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ENGINE EVALUATION (U).

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B-2707 PHASE III PROPOSAL DOCUMENTATION INDEX

1.0 SUMMARY

Throughout Phase II-C, as in earlier phases of the SST Program, The Boeing Company has maintained a program of design analysis and performance evaluation, with the coordinated participation of both engine contractors. This management approach was maintained to ensure that either of the engine offerings can be installed on the B-2707 and that the installed engine performance characteristics will closely match those of the B-2707 to produce near optimum airplane performance. As a result, the General Electric GE4/J5P powered B-2707 (GE) and the Pratt and Whitney Aircraft JTF17A-21B powered B-2707 (P&WA) airplanes are very competitive in terms of overall airplane performance criteria, and either would make an outstanding airline airplane.

Each engine has certain advantages and each has certain development risks.

The purpose of this document is to report the technical evaluation of the two engines. Considerations in this evaluation include engine/airplane matching and performance, engine design, installation compatibility, development plan, development risk, and growth potential.

A brief summary of the technical evaluation for each of the major areas follows.

1.1 ENGINE/AIRPLANE MATCH AND PERFORMANCE

The thrust characteristics of the two offered engines are such that, at the selected airflow sizes, both are matched at near optimum range-payload for the B-2707 airplane at 675,000 lb takeoff gross weight. However, the GE4/J5P turbojet has greater transonic thrust capability than the JTF17A-21B turbofan and therefore affords flexibility to meet sonic boom restrictions.

The range-payload performance of the B-2707 (GE) and B-2707 (P&WA) are essentially equal on a standard and hot day for the design mission. For missions requiring longer subsonic or shorter supersonic range increments, the P&WA turbofan engine has an advantage.

The differences in these overall performance considerations are small and final installation and performance matching of either engine could offset differences described here.

The estimated engine prices and development costs are very similar and the direct operating cost values calculated by the ATA formula are therefore essentially identical.

The GE4/J5P turbojet, at 620 pps SLS airflow, has better community noise characteristics than the JTF17A-21B at 687 pps, during takeoff and approach. The airport sideline noise levels for the two engines at full power are essentially equal at 117 PNdB while the takeoff community noise at three miles, after power cutback, are 96 PNdB and 105 PNdB for the GE and P&WA engines, respectively. Landing approach noise is 105 PNdB and 115 PNdB for the GE and P&WA engines, respectively, one mile from the runway threshold.

1.2 GROWTH POTENTIAL

The GE engine growth after 5 years of commercial service will improve the cruise SFC by about 3 percent, the takeoff thrust by 9 percent and the transonic thrust by 13 percent. Supersonic cruise airflow will increase about 9 percent which will require an inlet change but not an engine frame size change. All modifications suggested appear attainable. Thrust increases up to 42 percent are possible, through zero staging the compressor and increasing the engine frame size.

The P&WA engine growth after five years of commercial service will improve the cruise SFC by about 4 percent and improve the takeoff and transonic thrust by 12.5 percent. However, the compressor and fan modifications and other component improvement required to obtain this growth appear more difficult to attain. The transonic thrust increase of 12.5 percent, which is available by increasing transonic airflow through reduced solidity of compressor blading, will require an inlet size increase. This thrust increase requires a 10 percent increase in airflow and possibly an increase in the present compressor size.

However, a further design refinement of the P&WA engine by raising the bypass ratio looks promising. By increasing the bypass ratio to 1.6, additional takeoff and transonic thrust increases as high as 12.5 percent are possible without adding a turbine stage, making a total of 25 percent when combined with the previous improvements. Matched cruise SFC would be improved by 6 percent relative to initial production. Further thrust growth through increased bypass ratio is possible by adding a turbine stage.

It would appear that the thrust growth potential of the GE4/J5P is achievable with less engine redesign, whereas the P&WA JTF17A-21B with an increased bypass ratio could provide a better SFC growth. The engine weight changes occurring due to engine growth were not considered.

1.3 INLET/ENGINE COMPATIBILITY

Complete inlet/engine compatibility can be developed for the B-2707 airplane using either the GE4/J5P or the P&WA JTF17A-21B engines. The inlet incorporates features which provide wide stability margins for engine-generated disturbances, low circumferential distortion, and the means to adjust the moderate radial distortion to favor the particular engine selected.

From the standpoints of engine airflow stability, tolerance to distortion and dynamic interactions, the compatibility development would be the least difficult with the GE4/J5P engine. This is due to the basic cycle, design, and control concepts involved. From the standpoint of airflow matching, the P&WA JTF17A-21B has the advantage of lower subsonic inlet drag and a smaller bypass area requirement.

1.4 DEVELOPMENT RISK

The evaluation of the development risk of engine components has taken into account the performance, life, weight, and complexity. Judgment factors, which have been used, include design goals, demonstrated performance, past experience, technical capability, and the design approach.

In some instances, component design and demonstrated performance have shown that the component probably does not involve a major risk. For other components, an attempt has been made to categorize the degree of risk.

The major development risk categories have been classified by their SST implications as follows:

- Category 1: Increased development program cost
  Reduced parts life
  More complexity
  Minor program delays
- Category 2: Payload-range decrement
  Increased DOC
  Increased program delays
- Category 3: Major program delay
  Major program redirection

The risk evaluation results are listed in Table 1-A.

Table 1-A. Development Risk Summary

Item	GE	P&WA
Fan	-	1
Compressor	1	1
Fan + compressor	-	2
Main burner	-	-
Turbine performance	-	2
Turbine life	1	1
Augmentor	2	2
Nozzle performance	2	2
Thrust reverser	1	2
Weight	2	2
Controls and dynamics	1	3

The GE engine is a conventional afterburning turbojet engine which has as its technological base the GE J93 engine. The major development risk with the General Electric engine is the augmentation system, in particular the life of the high-temperature parts.

The P&WA duct burning turbofan is a new development in supersonic engines which entails development risks in several areas particularly the control system and associated engine dynamics.

Based on the information available at this time, the Boeing evaluation of these engines indicates that the risk in developing the GE4/J5P engine is lower than the risk with the JTF17A-21B engine.

1.5 INSTALLATION DIFFERENCES

The GE pod is 446 in. long, 89 in. maximum diameter, and weighs 14,312 lb. The P&WA pod is 345 in. long, 88 in. in diameter and weighs 13,865 lb.

One of the installation differences between the P&WA and GE engines is the reverse thrust available. The GE engine provides 50 percent of the maximum dry thrust or 23,500 lb in reverse as opposed to the P&WA 40 percent of maximum dry thrust or 14,080 lb. Under emergency conditions (icy runway) the stopping distance will be approximately 800 ft shorter for the GE-powered airplane.

Because of the higher moment of inertia of the GE rotor, the mount loads due to engine seizure are considerably greater on the GE engine.

The larger compressor blades of the GE engine provide a greater capacity to ingest ice, birds, etc. without damage.

The windmilling horsepower and rpm available from the P&WA turbofan are adequate to provide emergency power in case of fan engine failure. The GE turbojet does not provide adequate windmilling power and for the B-2707 (GE) a ram air turbine is provided.

The location of the rotating components of the GE engine is farther forward than with the P&WA engine. This restricts the location of fuel in the horizontal tail in a region above the compressor, thus decreasing the airplane fuel capacity by 5,628 lb.

The thermal environment of the nacelle in the compressor region during cruise is 950°F for the GE engine and 650°F for the P&WA engine. The engine accessories for the GE engine are mounted in an accessory capsule which isolates the accessories from the high nacelle temperatures. With the P&WA engine, the accessories are mounted in an exposed arrangement around the engine. Little difference exists in the maintainability of the two arrangements.

The GE engine is more accessible for inspection and maintenance of the interior of the engine than the P&WA turbofan.

1.6 MAINTENANCE, RELIABILITY, AND SAFETY

From a maintainability standpoint, the GE4/J5P engine is better than the P&WA JTF17A-21B engine. The maintenance frequencies do not appear to be drastically different. The on-airplane maintenance effort should be about equal except for internal inspections and onboard capability where the GE4/J5P has better provisions for internal inspection. The major difference between the engines is seen in the off-airplane repair and overhaul. The GE4/J5P, because of its modular construction, should have a lower elapsed time and cost for repair and overhaul.

The reliability evaluation considered basic engine design reliability, goals and apportionments, component life goals considering the severity of operating conditions, and the estimated effectiveness of the two manufacturers' reliability programs. In all but the last category P&WA appears to be slightly better than GE, and with regard to reliability programs they are essentially equal.

In the area of safety, the PWA engine offers some installation advantages.

1.7 DEVELOPMENT PLAN, SCHEDULES AND FACILITIES

The development program proposed by the two engine companies is considered adequate in both the component and engine development testing. Endurance testing is emphasized. Pratt and Whitney Aircraft and GE have provided adequate engine test facilities to accomplish their programs. Both companies plan to conduct the inlet/engine compatibility testing at AEDC, but P&WA plans to perform the performance demonstrations in their own facility while GE will use AEDC facilities for this purpose. Both engine companies can meet Boeing prototype and production engine delivery requirements.

2.0 ENGINE DESCRIPTION

2.1 ENGINE AND COMPONENT DESCRIPTION

The information and description presented in this document represent information available to Boeing as of August 8, 1966. This includes information obtained during a visit to the two engine manufacturers in mid-July 1966, and also data from preliminary drafts of parts of the engine manufacturers' proposal documents. It should be noted that some differences were found between the early data obtained in July 1966, and that information obtained in the draft proposals.

2.2 GENERAL ELECTRIC ENGINE GE4/J5P

The General Electric engine is a single spool, 620 lb/sec sea level static airflow turbojet engine with afterburning thrust augmentation. The design pressure ratio is 12.3, with takeoff and climb turbine inlet temperature of 2,250°F and cruise turbine inlet temperature of 2,200°F.

2.2.1 Compressor

The compressor is a low aspect ratio, nine stage, axial flow design. The inlet guide vanes (IGV) and first stator row are variable for engine starting and low speed acceleration, while the last six stator rows are variable for increased surge margin in cruise, and engine-inlet flow matching purposes. The last stator can be actuated as a windmill brake in the event of in-flight engine shutdown. Inlet flow distortion effects are stated to be minimized by reduced loading of the front stages, low aspect ratio blades, and the use of a variable exhaust nozzle under all operating conditions. Hollow compressor blades are used to reduce compressor weight. Rotor blades can be replaced in moment-weighted pairs without rotor disassembly or rebalancing, while stator vanes are individually replaceable.

2.2.2 Main Burner

The annular combustor is similar to past GE burner designs. Fuel is provided for the combustion process by 42 variable area dual orifice fuel nozzles. The liner wall is film cooled.

2.2.3 Turbine

The GE4/J5P has a 2-stage, axial flow, air-cooled turbine, designed for operation at a high gas temperature for extremely long duration. Cooling air from the compressor, including compressor discharge air flowing around the combustor and sixth stage bleed air through the rotor shaft, provides film and convection cooling for turbine vanes, blades, disks, and second stage shroud. Turbine vanes are individually replaceable and blades are replaceable in moment-weighted pairs.

2.2.4 Augmentor

The thrust augmentor for the General Electric engine is a fully modulating afterburner having two stages of fuel injection and incorporating four gutter flame holders. Spark ignition is provided only during the initiation of afterburner operation. The afterburner liner is film cooled by turbine discharge gas.

2.2.5 Exhaust Nozzle

The exhaust nozzle is a two stage, blow-in door ejector design with an actuated primary or convergent section for variable throat area, and an aerodynamically positioned secondary or divergent section for variable exit area. Nozzle cooling is provided by compressor bleed air, inlet secondary air, and turbine exit air.

2.2.6 Thrust Reverser

This unit reverses thrust by moving the primary or convergent part of the exhaust nozzle aft and inward, thereby blocking the rearward flow of exhaust gas, causing the gas to escape in the forward direction through reverser cascade openings in the tailpipe wall. This action provides 50 percent of maximum dry thrust as reverse thrust.

2.2.7 Bearings and Seals

The bearing arrangement for this single spool engine consists of a single ball type main thrust bearing mounted on the mid-frame and interchangeable roller bearings on the front and rear frames. Seals are pressurized, floating face, carbon type with windback labyrinth seals as backup.

2.2.8 Control System

The engine is controlled by positioning a single

power lever for reverse, idle, maximum dry, maximum augmentation, and all intermediate conditions. The control system also positions the compressor variable stators, the variable exhaust nozzle, and the thrust reverser.

2.2.9 Materials
The materials selected for this engine include some which have been used extensively in subsonic engines, and also a high percentage of age-hardened nickel and iron based alloys, and titanium alloys which have not been used extensively in past engines. The forward part of the compressor uses titanium alloys, while the high temperature rear half uses nickel-based alloys as does the hot turbine section. The basic structure of the engine is of nickel or iron-based sheet metal components, with the exception of the compressor front frame and support vanes, which are of titanium alloy. Highly stressed parts are of Inconel 718, Rene 41 and Rene 62, while lower stressed parts such as the combustor are of Hastelloy-X or modifications thereof.

2.3 PRATT AND WHITNEY AIRCRAFT ENGINE JTF17A-21B
The Pratt & Whitney engine is a dual spool turbofan engine of 687 lb/sec sea level static airflow, with fan duct burning for thrust augmentation. The sea level static (SLS) bypass ratio is 1.31. Takeoff and climb turbine entry temperature is 2300°F while cruise temperature is 2200°F. Overall engine pressure ratio is 13.0, with a duct fan pressure ratio of 2.96 when operating at SLS conditions.

2.3.1 Fan and Compressor
The fan is a two stage transonic design. Two vibration dampers are used on each fan rotor. Exit flow from the fan is divided into secondary or duct flow, and primary flow, between the second rotor and second stator. Inlet guide vanes (IGV) are not used in front of the fan, but a three-position IGV is located in front of the high pressure rotor. This variable IGV serves as the windmill brake for the engine during inflight engine shutdown.

The high pressure compressor, driven through a concentric outer shaft, is a six stage axial compressor having a SLS design pressure ratio of 4.84. The high pressure compressor is of short chord design in the forward stages.

2.3.2 Main Burner
The annular combustor, termed a ram induction burner, features a low level of diffusion of compressor discharge gas, and a corresponding short length. Liner cooling is accomplished by high velocity convective flow over the outer surface of the burner liner. Ignition is provided by dual igniters, while fuel injection is accomplished through 24 dual orifice, variable secondary area nozzles.

2.3.3 Turbine
The turbine section consists of a single-stage high pressure turbine driving the high pressure compressor, and a two-stage low pressure turbine driving the fan. Cooling of all three vane rows, and the first two blade rows is accomplished by compressor discharge air employing convection methods, including impingement cooling of leading edges. Material selected for the turbine includes PWA 658 for all blades, and the last two vane rows, and TD nickel for the first vane. Vanes are individually replaceable and blades are replaceable in moment-weighted pairs.

2.3.4 Augmentor
The augmentor, which is the first fan duct heater to be applied to any aircraft, is a unique component in this engine. Major features include a relatively low level of diffusion between the fan and the ram induction augmentor. Two zones of fuel injection are provided; the first is a ram induction burner with the same nozzles as the main burner, and a second zone, downstream of the first, provides fuel injection and mixing into the upstream flame zone. Spark ignition is provided for the forward zone of combustion, while the second zone is autoignited from the upstream zone. The liner walls are film cooled.

2.3.5 Exhaust Nozzle
The exhaust nozzle is a convergent-divergent blow-in door ejector-reverser combination. The convergent nozzle of the primary or core engine is of fixed area. The fan duct convergent nozzle is of variable area. Secondary air from the inlet is used to convectively cool the duct convergent nozzle and ejector. Tertiary air is automatically provided through a pressure balanced blow-in door system for nozzle performance at transonic and subsonic speeds. The reverser clamshells have two non-reverse positions, fixed by the tertiary door location. For flight speeds above Mach 1.2, the clamshells form part of the

divergent flow path for both primary and duct flow streams. At Mach numbers less than 1.2, the clamshells move to a slightly convergent position, admitting the tertiary air to the nozzle section. The nozzle exit flaps are pressure located and act as a convergent or divergent part of the nozzle for both streams, depending upon the nozzle pressure ratio.

2.3.6 Thrust Reverser

Reverse thrust is obtained by rotating the clamshell doors, which are integral parts of the exhaust nozzle system, to the closed position. This location of the clamshells partially blocks the rearward escape of the exhaust gases and deflects the gas to a forward direction through the tertiary door openings or "blow-in-doors" in the nozzle outer shell. The failsafe position of the reverser clamshells is in the open or forward thrust position. The reverse thrust goal is 40 percent of maximum dry forward thrust.

2.3.7 Bearings and Seals

The two spools of this engine are designed with four bearings. This concept is a departure from existing two spool engines in service today. The No. 1 bearing, the thrust bearing for the low pressure or fan spool, is a ball bearing mounted near the second fan rotor. The No. 4 bearing, also on the low pressure shaft, is a roller type, mounted at the last stage of the low pressure turbine. The high pressure rotor bearings are No. 2, the thrust bearing located at the high pressure compressor IGV, and No. 3, again a roller bearing, forward of the high pressure turbine. Seals are hydrostatic with tandem pressurized and vented labyrinth seals as backup. Fan discharge air is used to pressurize the labyrinth seal compartment, which is then vented to ambient.

2.3.8 Controls

The engine is controlled by positioning a single power lever for reverse, idle, maximum dry, maximum augmentation, and all intermediate conditions. The control system also positions the compressor variable IGV, the variable duct exhaust nozzle and the thrust reverser.

2.3.9 Materials

A more extensive selection of age hardened iron, nickel based alloys, and titanium alloys has been made for this engine than in past engines. The fan section is of titanium alloy forgings while the high temperature compressor section is of titanium, and iron and nickel based alloys. Hastelloy-X is used in both the main and duct burners.

3.0 PERFORMANCE, NOISE, AND ECONOMICS

The maximum range/payload objective for the B-2707 is obtained with both GE and P&WA engines at the selected sizes with acceptable takeoff field length, airport and community noise levels, and transonic thrust margin. However, the degree of acceptability varies with the engine choice.

The GE4/J5P turbojet can meet the B-2707 takeoff requirements, and provides acceptable transonic performance and has an additional takeoff thrust capacity for payload growth.

The JTF17A-21B turbofan has less than desired transonic thrust margin needed for operating flexibility and contingencies.

The Boeing estimated noise levels, based on engine manufacturers' and Boeing test data, indicates that either engine will meet FAA takeoff airport and community noise objectives although the GE engine has somewhat more flexibility to trade off airport and community noise by adjusting airplane takeoff procedures. The landing noise levels will be lower than the FAA objective in the case of the GE engine, and higher in the case of P&WA.

The P&WA engine provides better all subsonic range. The single engine out range is about the same for either engine, and both engines enable reaching a destination on the most critical legs.

Engine price, development costs, including the dependent maintenance, insurance and depreciation cost factors, and trip fuel are approximately the same for both engines and consequently the direct operating cost for both engines should be essentially equal.

3.1 ENGINE SIZE

The engine size selection for the GE and P&WA engines was based on achieving maximum range/payload while providing satisfactory airport and community noise levels and meeting the takeoff field length and transonic thrust requirements on standard and hot days.

Engine size effects on range and transonic thrust margin are shown on Fig. 3-1. Both engines give peak range for airplane gross weight of 675,000 lb at ΔP_{MAX} = 2.5 psf. The GE engine at 620 lb/sec has adequate thrust margin. The transonic thrust margin at ΔP_{MAX}=2.5 psf for the desired P&WA is marginal (0.23 versus 0.3 desired on standard day). The .3 transonic thrust margin is considered the minimum desired to allow for contingencies that can arise during airplane acceleration, such as nonoptimum climb path, nonstandard day, and unanticipated drag increase, or thrust deficiency.

At the primary and duct burner temperature levels, increased maximum transonic thrust capability in the JTF17A-21B cycle cannot be achieved without advancing the state of the art in turbine inlet and duct heater temperatures. Increased transonic thrust by increased airflow is difficult to attain, due to mismatch of transonic and supersonic cruise airflow for a fixed primary nozzle design. Use of a variable primary nozzle for higher transonic airflow capacity, and afterburning in the primary stream have been considered, but have not provided a significant thrust increase considering the increased weight and complexity of cooling and control of the primary nozzle. Increased transonic thrust margin appears available only by increased transonic airflow by means of engine scaling or higher bypass ratio.

3.2 RANGE AND PAYLOAD

The range and payload capability of the two engines for standard and hot days is shown on Fig. 3-2 (International) and Fig. 3-3 (domestic). The performance of the two engines is almost

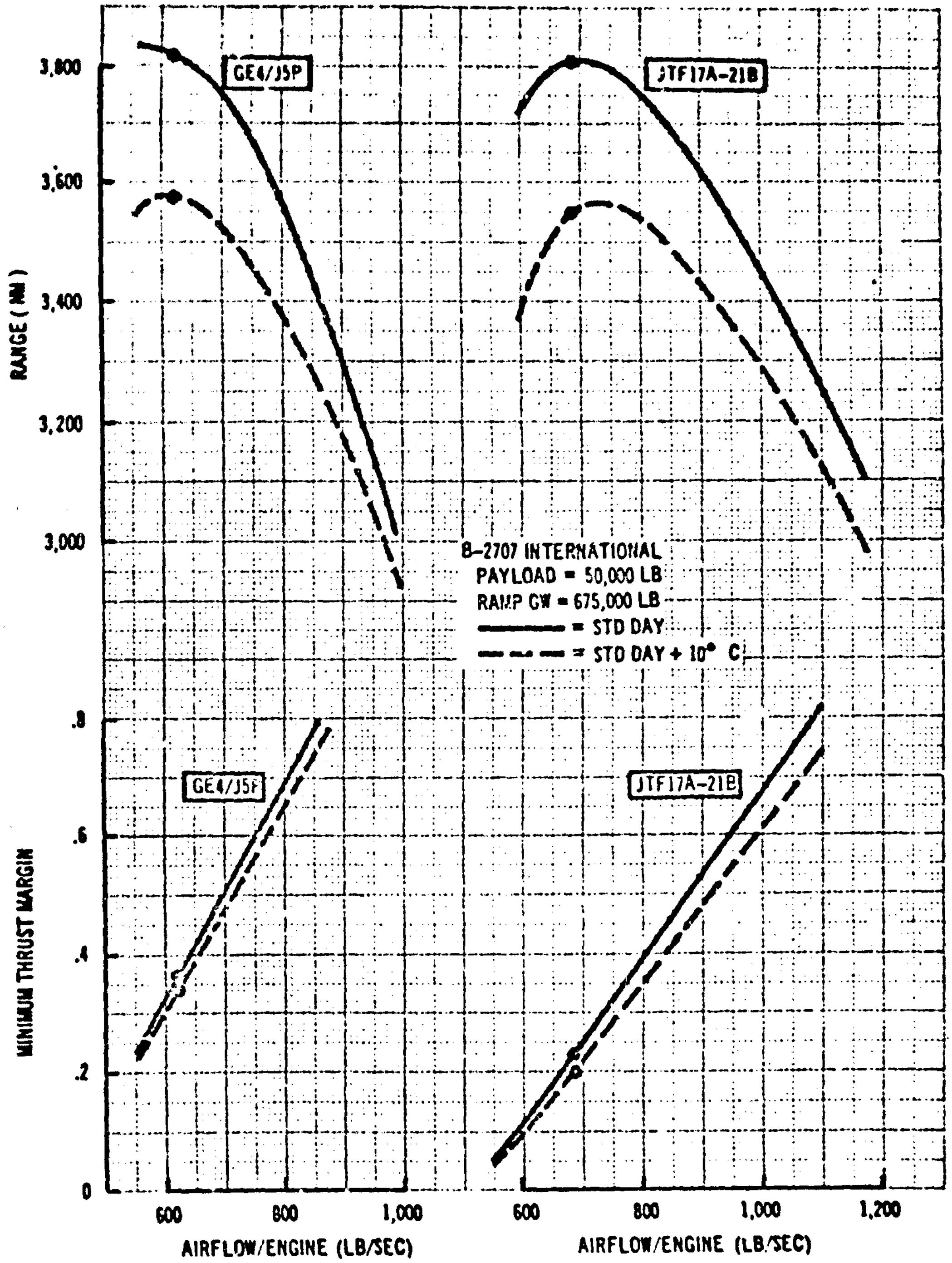


Figure 3-1. Model B-2707 Airflow Sizing, $\Delta P_{max} = 2.5$ PSF

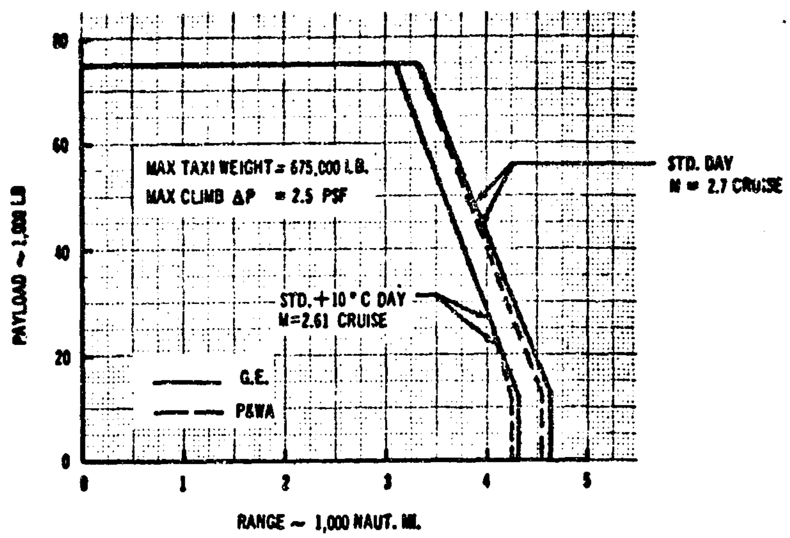


Figure 3-2. Payload - Range. International

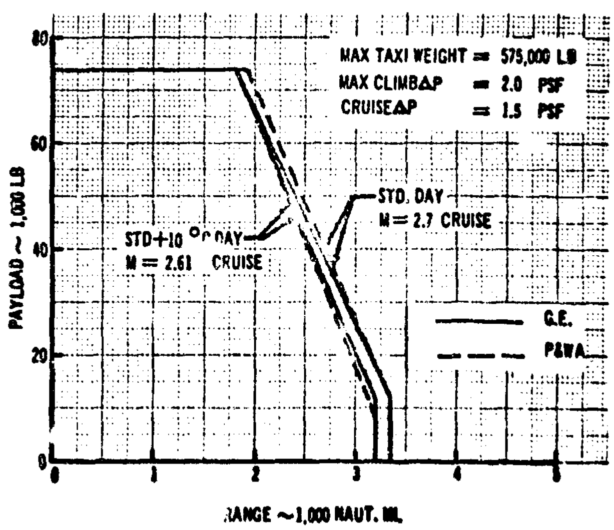


Figure 3-3. Payload - Range. Domestic

identical for standard and hot days with both airplanes.

The engine supersonic and subsonic cruise performance and match points are shown on Fig. 3-4. The engine climb performance comparison is shown on Fig. 3-5. The turbojet has lower supersonic cruise and climb Specific Fuel Consumptions (SFC), but higher subsonic SFC's. The net result is that the range and payload performance is the same for either engine for the basic mission.

The effect of increasing the subsonic leg at the beginning or end of the mission is shown on Fig. 3-6. The P&WA turbofan has 664 nautical miles greater range for an all subsonic mission.

3.3 FUEL CONSUMPTION AND DISTANCE

The fuel consumed and distance covered for the B-2707 (GE) and B-2707 (P&WA) for standard day are listed in Table 3-A.

Due to the relative SFC differences between the turbojet and turbofan cycles at the required thrust, the B-2707 (GE) requires more reserve fuel weight, while the B-2707 (P&WA) consumes more fuel during climb and acceleration. Trip fuel which is an important factor in direct operating costs is approximately the same for either engine.

3.4 ENGINE-OUT PERFORMANCE

The effect of one and two engine-out on airplane range and reserve fuel consumed is shown in Fig. 3-7.

The B-2707 with either engine can achieve 3,470-nmi range with one engine out at mid-range point. Both engines require off-loaded payload to achieve 3,470 nmi with two engines out.

The effect of one augmentor inoperative for the B-2707 (GE) and B-2707 (P&WA) is shown in Fig. 3-8 for a standard day. A greater amount of reserve fuel is consumed by the turbofan since the three remaining engines must make up a greater amount of thrust.

3.5 AIRPORT AND COMMUNITY NOISE

The noise produced by the engines on the international and domestic versions are shown on Figs. 3-9 and 3-10, and are based on engine manufacturers quoted suppression and Boeing estimates for choked inlet noise suppression. Table 3-B summarizes the noise suppression values used during takeoff and landing calculations.

Table 3-A. Model B-2707 Fuel Consumption and Distance — Standard Day

Item	B-2707 (GE)		B-2707 (P&WA)	
	Fuel (lb)	Distance (nmi)	Fuel (lb)	Distance (nmi)
Takeoff	8,210		7,265	
Climb	90,190	342	103,440	432
Mach 2.7 Cruise	186,791	3271	181,626	3180
Descent	5,270	206	4,304	196
Reserves	47,039		42,605	
Trip Fuel	290,461		296,635	

Table 3-B Model B-2707 Total Noise Suppression

	B-2707 (GE)	B-2707 (P&WA)
Takeoff Max Augmented, PNdB	4	4
Cutback at 18,000 lbs thrust, PNdB	7	9
Approach at 14,000 lbs thrust, PNdB	20	10

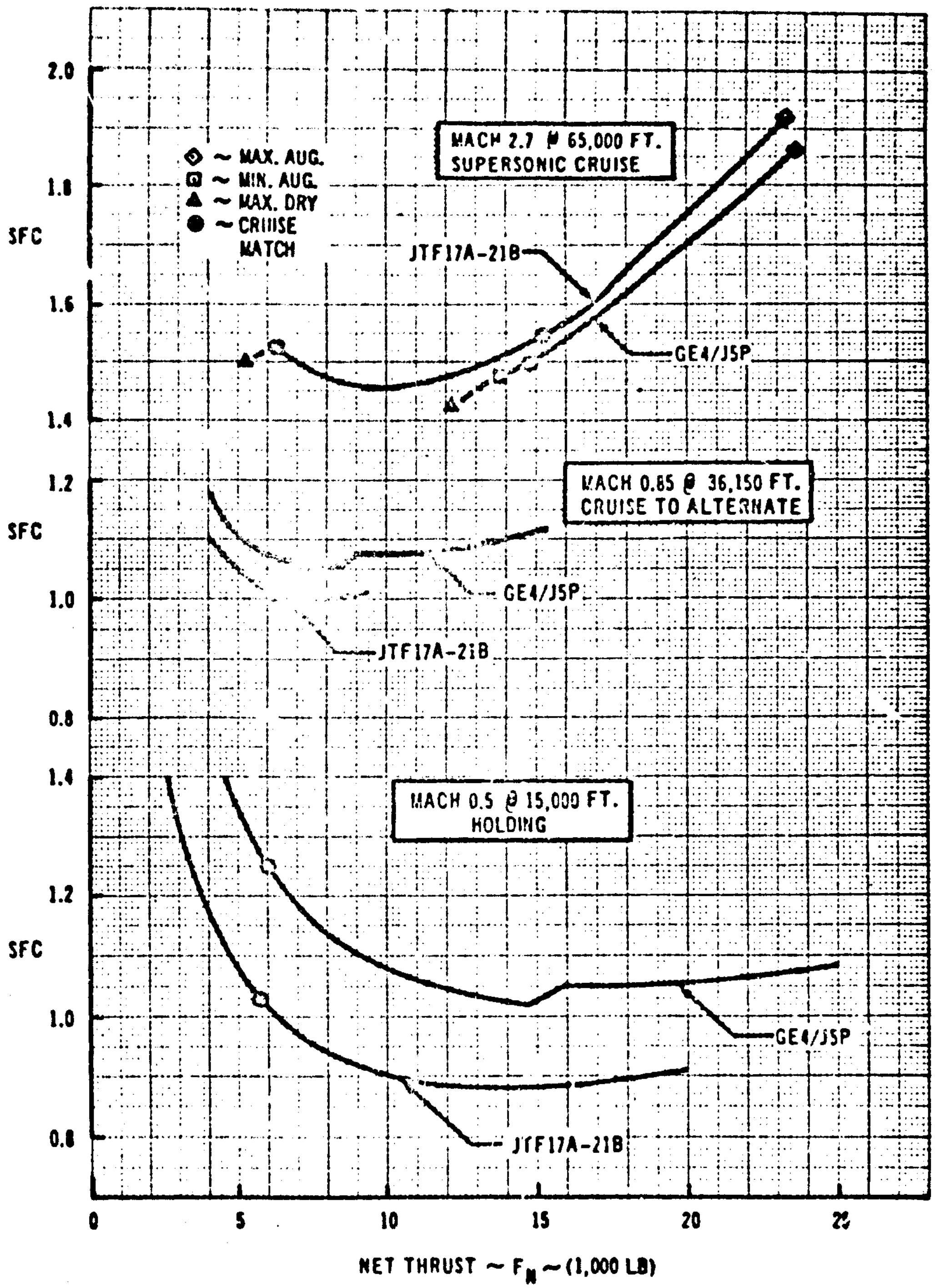


Figure 3-4. Supersonic and Subsonic Performance – Standard Day

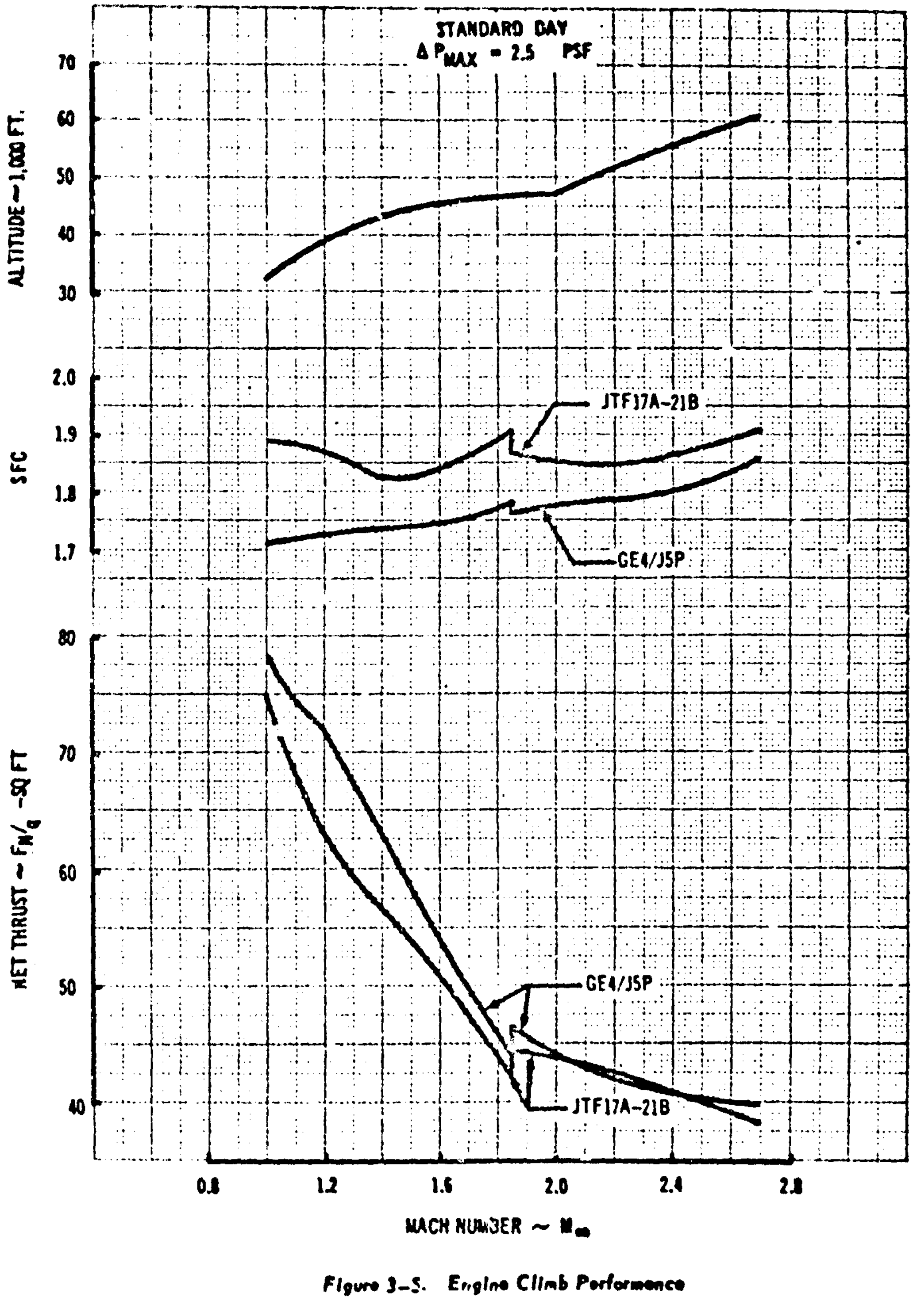


Figure 3-5. Engine Climb Performance

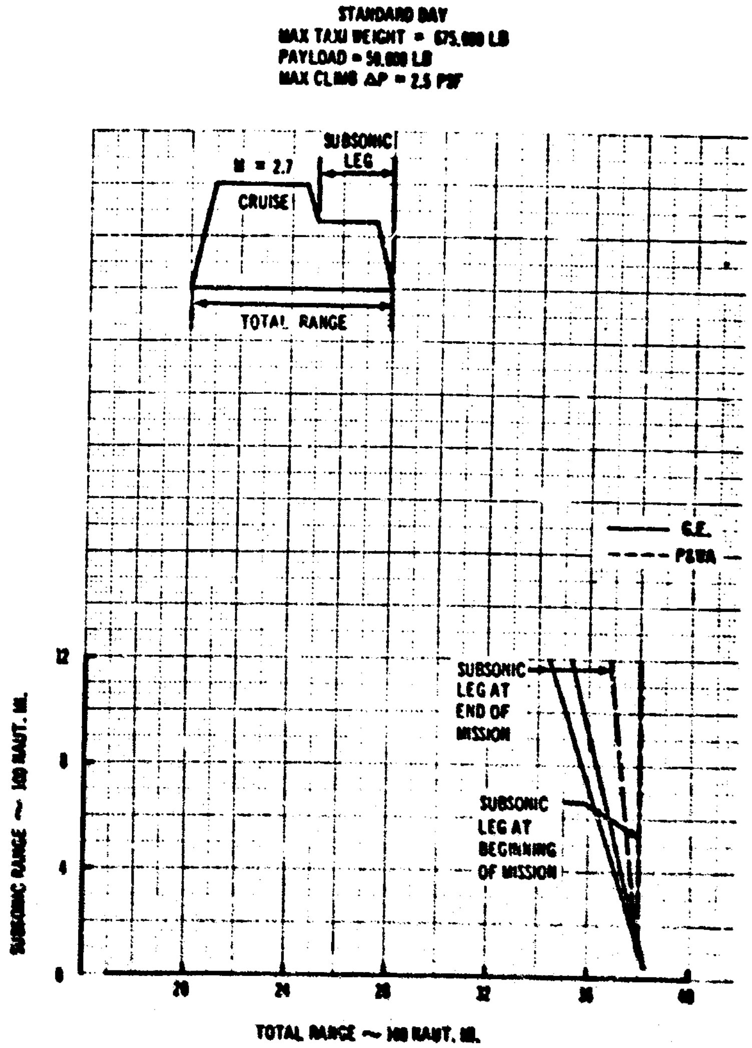


Figure 3-6. Subsonic Leg Performance

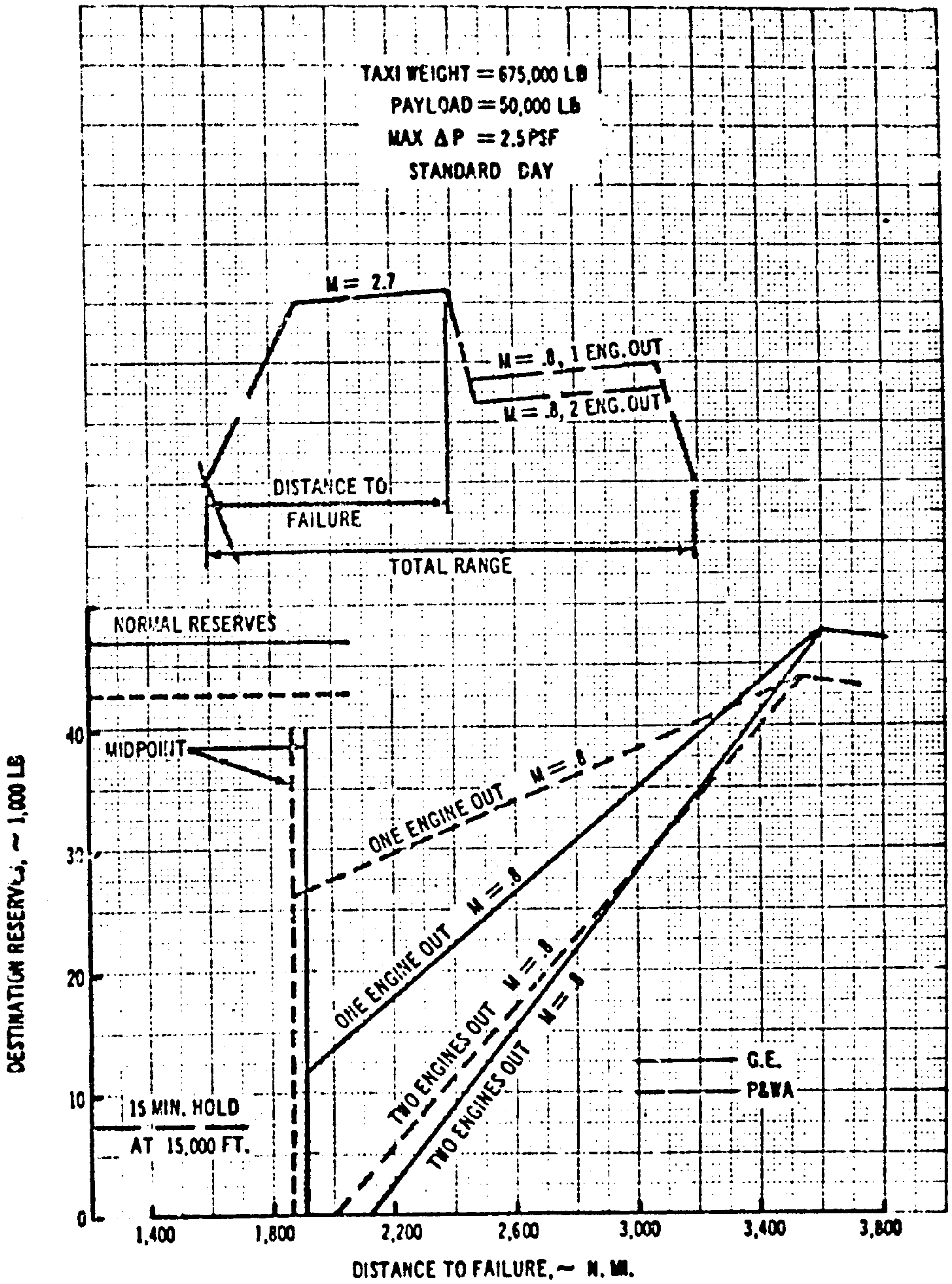


Figure 3-7. Range Capability With Engine Out

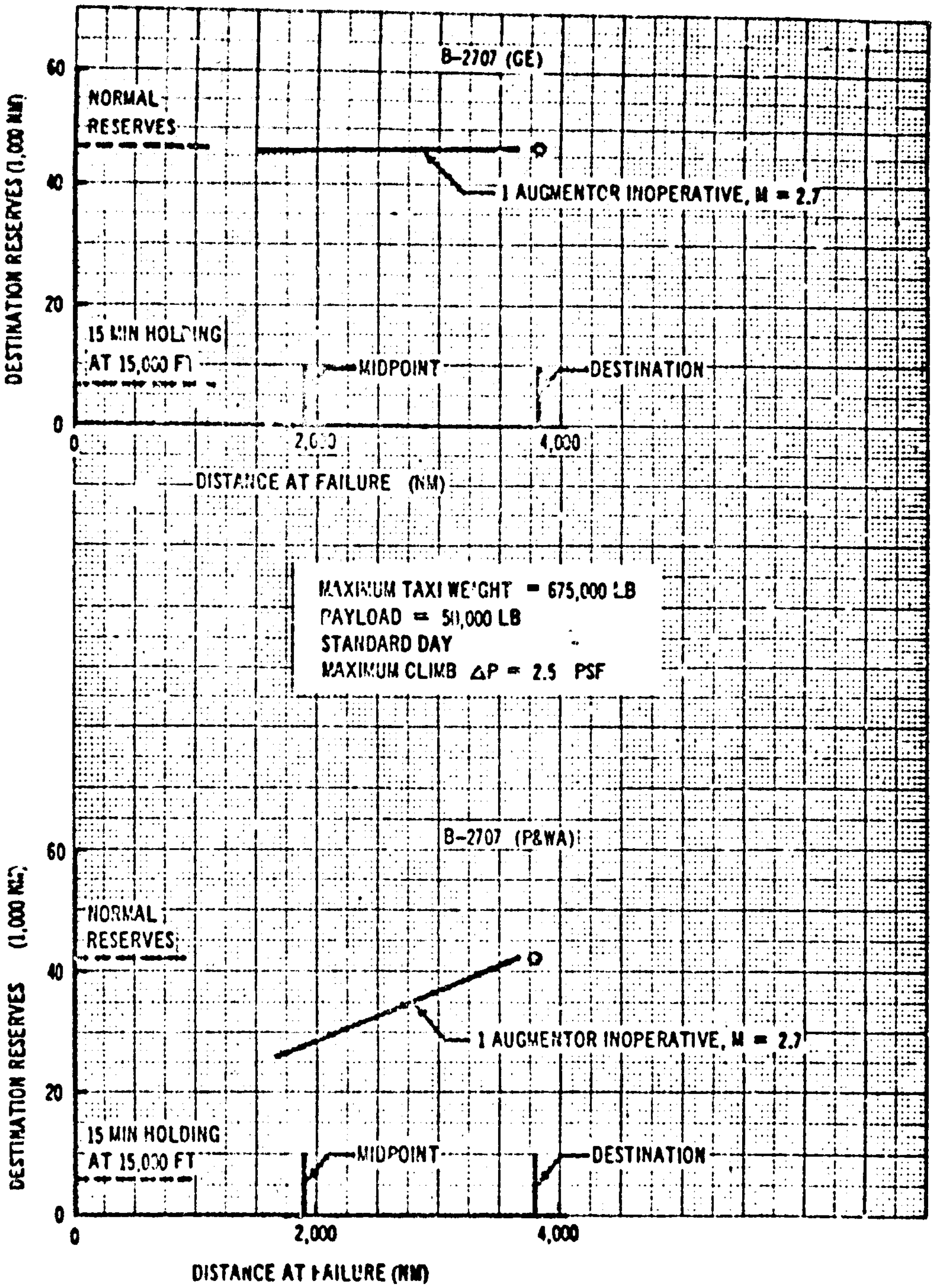
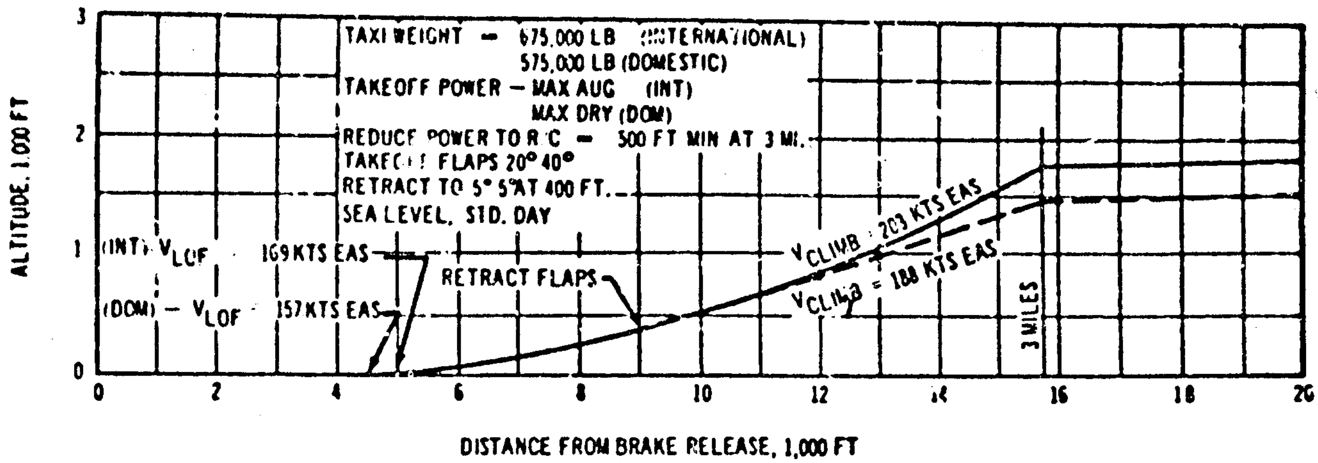


Figure 3-8. Range Capability With Augmentor Failure, Std Day



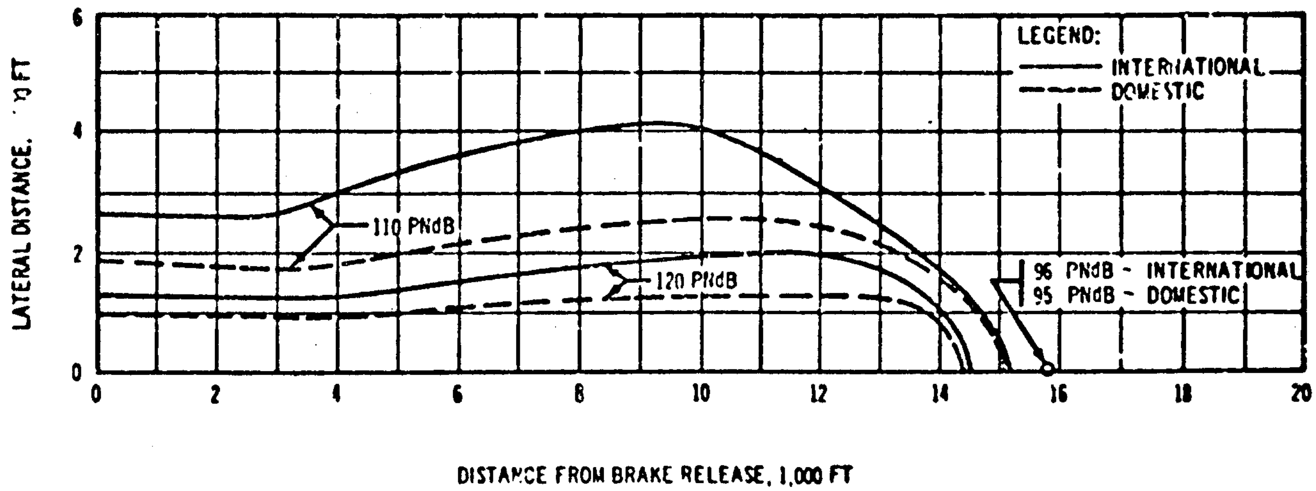


Figure 3-9. Model B-2707 (GE) Takeoff Performance & Noise Contours

V4-B2707-14

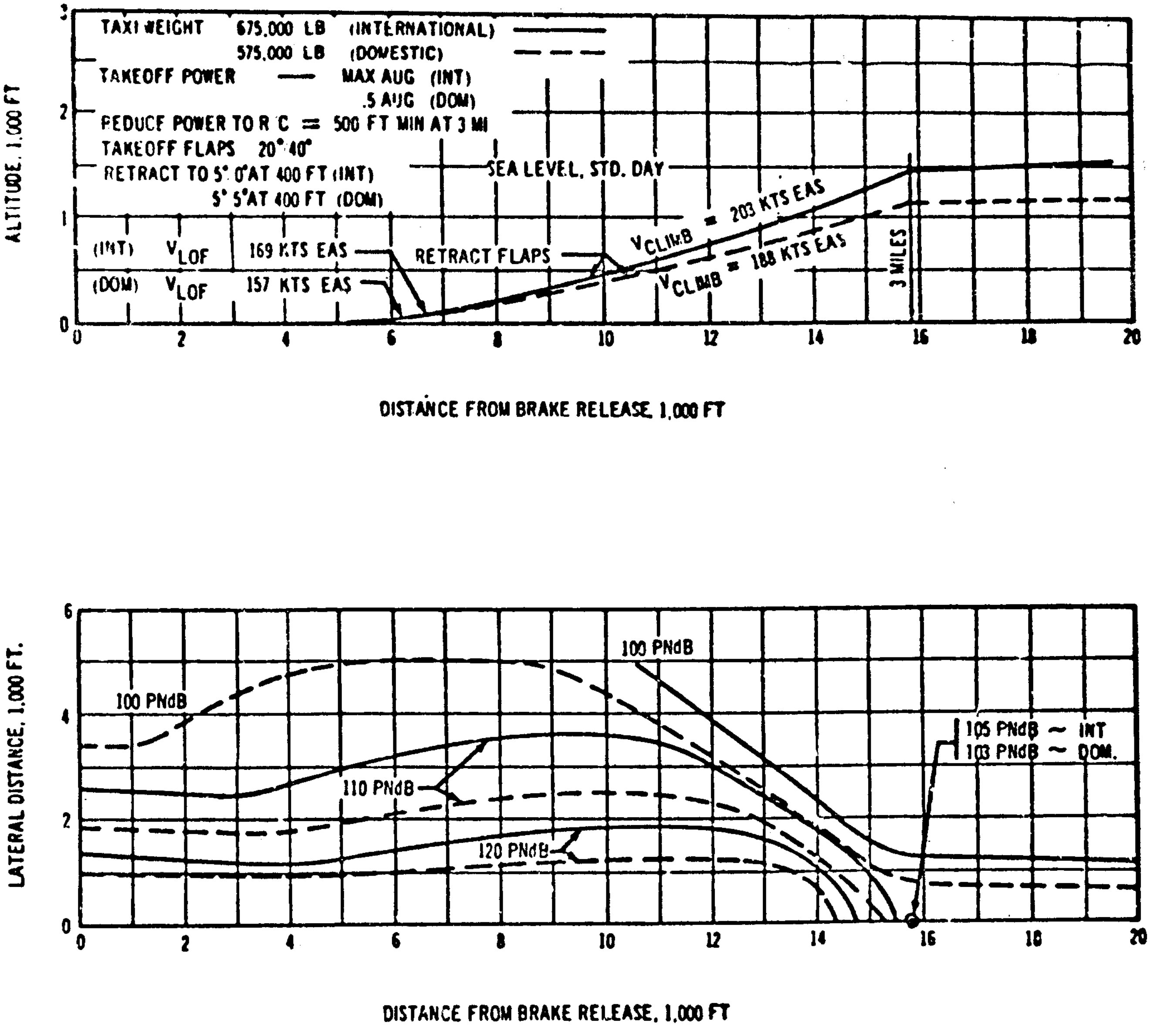


Figure 3-10. Model B-2707 (P&WA) Takeoff Performance & Noise Contours

Takeoff noise contours in Figs. 3-9 and 3-10 are based on the use of maximum augmentation with gradual thrust cutback over the community near the three mile point to achieve an unaccelerated 500 ft/min rate of climb.

The curve shows the PNdB noise contours at ground level and the flight path of the airplane. The higher takeoff thrust at maximum augmentation for the GE4/J5P engine results in a higher altitude over the community at the three mile point and nine PNdB lower community noise after cutback than with the P&WA engine. Takeoffs at less than maximum augmentation will result in lower noise levels in the vicinity of the airport with some increase in community noise levels. The trades between community noise, airport noise, and takeoff field length for reduced power settings are shown on Fig. 3-11 for both engines. As shown on the curves, the B-2707 (GE) at reduced augmented power can meet the FAA noise objectives. The B-2707 (P&WA) cannot achieve both FAA objectives simultaneously.

Approach noise contours for the two engines are shown on Fig. 3-12. The B-2707 (GE) value is 105 PNdB which meets the FAA objective of 109 PNdB. The B-2707 (P&WA) approach noise i 115 PNdB.

Effective jet noi suppression for high takeoff thrust levels and open nozzle control which reduces jet noise during cutback and approach power in combination with inlet choking for compressor noise suppression enable the B-2707 (GE) to meet the airport and community noise objectives. The same techniques have been applied to the turbofan. However, jet suppression is less effective for the high velocity primary stream and with present knowledge fan duct treatment cannot remove all of the fan noise exiting from the secondary nozzle.

3.6 ECONOMIC CONSIDERATIONS

3.6.1 Engine Price

The estimated production engine prices for the proposed engines are as follows:

GE4/J5P	$1,175,000
JTF17A-21B	$1,210,000

The engine manufacturer estimated development costs for each engine are:

General Electric	$607 million
Pratt & Whitney	$663 million

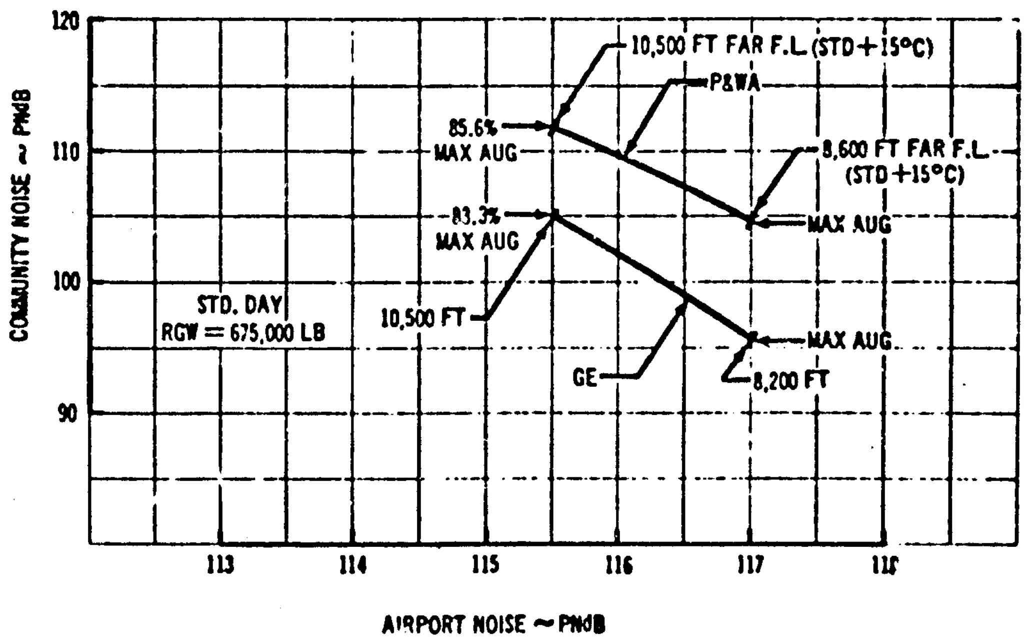
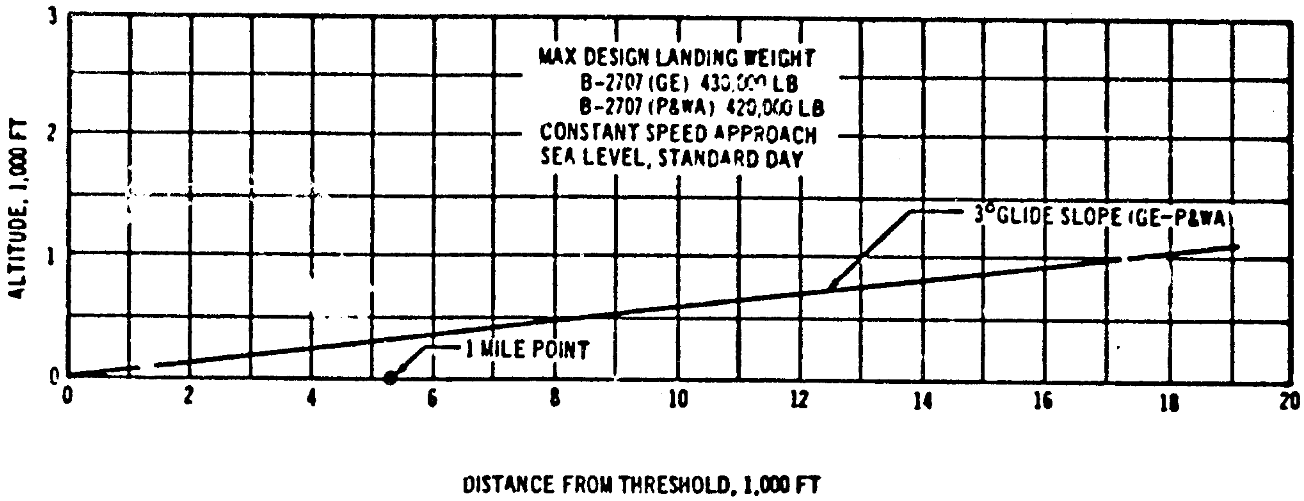


Figure 3-11. Airport & Community Noise Trades



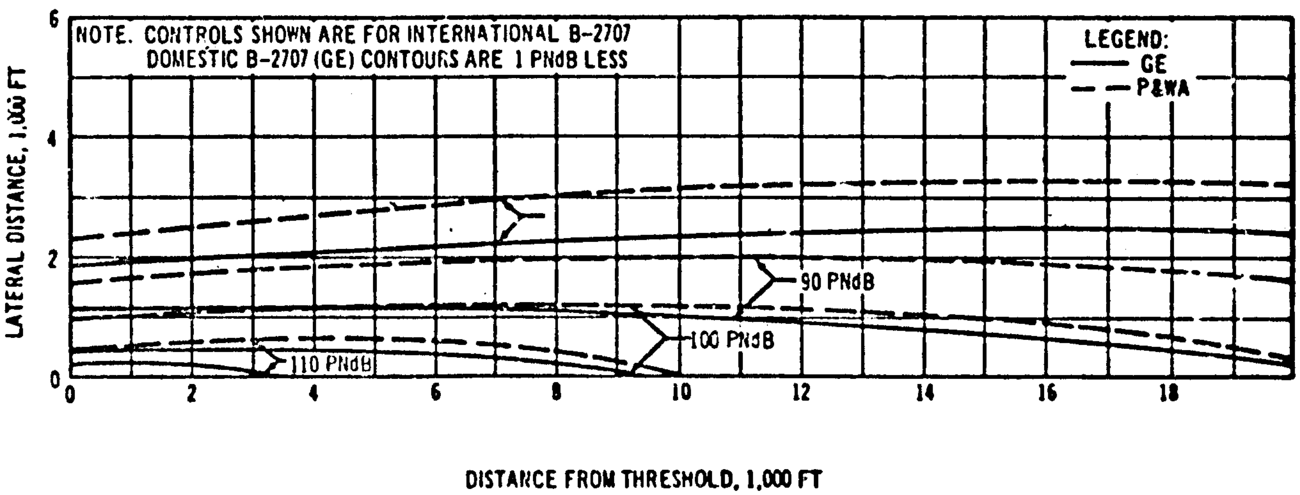


Figure 3-12. Perceived Noise Level Contours for Landing Approach

3.6.2 Direct Operating Cost

Direct Operating Cost (DOC) comparisons for the Boeing B-2707 International Airplane with with the proposed engines is shown on Fig. 3-13. The DOC's for the two engines are essentially equal at design range.

Direct OperatingCosts were determined using the Modified 1960 Air Transport Association method as specified in the SST Economic Model Ground Rules dated June 30, 1966 and are based on 3,000 hours annual utilization, 15-year depreciation period, 3,000 hours engine time between overhauls, 50 percent engine spares, 12 cents per gallon fuel price and an engine spare parts factor of 1.5.

Table 3-C lists a breakdown of the major factors which contribute to total DOC on the basis of cost per block hours.

The maintenance costs of the P&WA powered airplane are higher since in the ATA formula engine maintenance costs are computed as a percentage of engine price. The higher engine price and engine development cost amortization of the P&WA engine also results in higher insurance and depreciation changes for the airplane.

Table 3-C. Analysis of Direct Operating Cost Components

		GE4/J5P	JTF17A-21B
Unit engine price		$1,175,000	$1,210,000
Total DOC (to nearest dollar)	$/Block-hr	3520	3476
Sub-Totals fuel & oil at 2000 Stat. mi.	$/Block-hr	1385	1325
Maintenance	$/Block-hr	663	665
Insurance	$/Block-hr	355	357
Depreciation	$/Block-hr	919	929
Crew	$/Block-hr	200	200

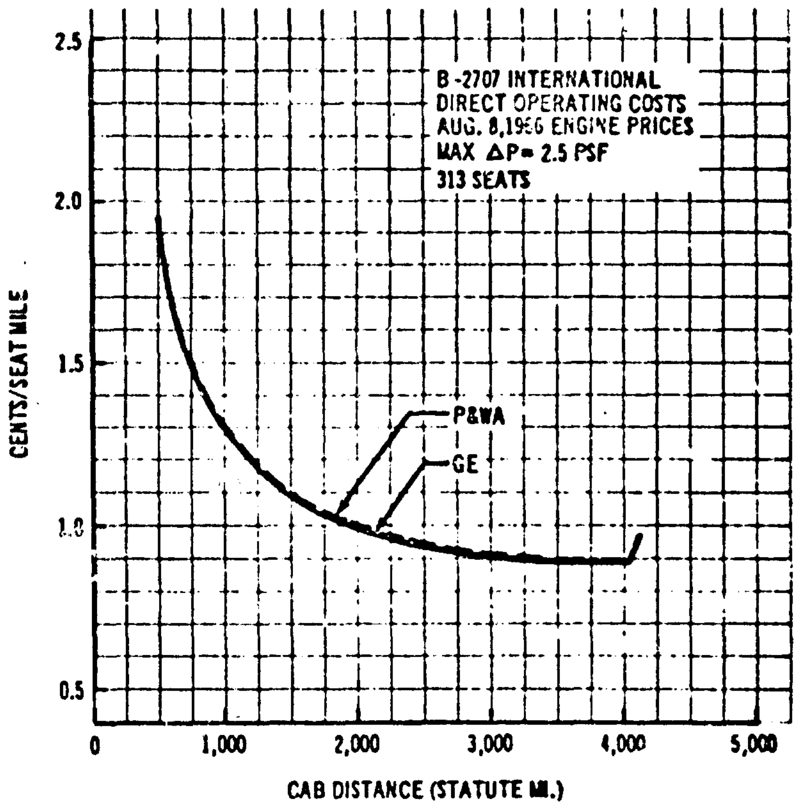


Figure 3-13. Direct Operating Cost Comparison

4.0 ENGINE GROWTH

Engine growth can come either in the form of thrust or SFC improvements in any portion of the flight spectrum. The performance of growth versions of the offered turbojet and turbofan depend on extensions to the operating temperature limits, component efficiencies, and/or flow capacity. Those growth items which can be accomplished without major modifications to the installation or scaling of the inlet are considered most attractive. Both engine companies offered some growth of this type. That growth which comes from large increases in airflow, would necessarily require major propulsion system redesign. A large thrust growth would have to be planned into aircraft production model changes, such as extended body and/or enlarged wing area versions, to be practically adopted, because of costs and development time involved in accepting these changes. However, the major engine rework developments which would call for new propulsion pod designs should be compared with direct engine and propulsion system scaling before they should be attempted.

4.1 GENERAL ELECTRIC ENGINE

The GE4/J5P engine was offered with a specific series of growth versions, which are made available at three and five years after initial commercial service. As shown on Table 4-A, the emphasis is on changes which increase thrust across the board. Four possible increased airflow versions are offered. Step No. 1, a 4.5 percent airflow increase at cruise, could probably be accommodated on the present aircraft pod design with a minimum of additional effort on Boeing's part. Step No. 2 requires an inlet change whereas Step No. 3 and 4 require both an inlet change and a change in pod and frame size. Cycle temperature increases, as noted on Table 4-A, are offered for each of these choices. Table 4-B shows the thrusts and SFC of the offered engine growth ratings.

The turbine temperature increase of 100°F would allow the present airplane to cruise at an SFC reduction of about 1 percent. The added thrust would not appreciably help the aircraft range factor as the aircraft is presently flying close to maximum L/D at this engine size. The increased airflow offered in step 3 yields an additional 9 percent in cruise thrust. The airflow improvement, coupled with the temperature increase would provide an equivalent SFC improvement of 3 percent (100 miles range improvement). Step No. 4 should offer a somewhat better situation, but no data was supplied on cruise thrust.

Although the range improvement on the present airplane is small, the gross weight improvement possibilities are quite substantial based on the 42.7 percent takeoff growth (Table 4-B) offered in Step No. 4, should this be preferred to scaling of the engine. Step No. 3 thrust increase of 20.9 percent provides considerable aircraft growth possibilities without consideration of engine scaling.

All modifications offered are believed to be normal developments in engine program.

4.2 PRATT AND WHITNEY AIRCRAFT ENGINE

The JTF17A-21B was offered with a five year growth in thrust of 12.5 percent at takeoff and transonic conditions, and matched cruise SFC improvement of 4.2 percent. These improvements would come about through increased airflow and component developments. Table 4-A lists the improvements to be expected in that time period. Table 4-B shows the thrust and specific fuel consumption (SFC) corresponding to these changes. The P&WA engine growth will depend on aerodynamic improvements in the compressor and nozzle, as well as cycle temperature increase.

Attainment of certain of these growth items appears to be more difficult than on the GE engine. Boeing cycle studies indicate that 12.5 percent takeoff thrust improvement (listed in Table 4-B) would require about 10 percent increase in airflow coupled with the component changes offered. The P&WA plan to achieve this

through reduced form and compressor blade solidity, appears optimistic. Pratt and Whitney Aircraft plan to improve the compressor efficiency by up to six percent as part of this program. Although the present design of 4.8 pressure ratio in six stages at 86 percent efficiency is within the present state of art, such an increase calls for a substantial improvement in this art over the next five years.

As a further development, P&WA offered to increase the bypass ratio of the engine from 1.3 to 1.6 without increasing the number of turbine stages, but a nozzle redesign is required and this would probably be accomplished by adding a compressor stage at the compressor inlet. The Sea Level Static (SLS) thrust is increased 25 percent, the transonic thrust by 30 percent and the SFC is reduced by a total of 6.2 percent from the basic engine offering. This evolution appears to be a reasonable method of thrust growth for this turbofan engine.

Both of the thrust growth methods proposed by P&WA would require a change in inlet size. The four percent cruise SFC improvement could be provided in the present propulsion installation.

4.3 SUMMARY

The GE engine could be modified to improve the cruise SFC by about two percent without requiring an inlet size change. A cruise SFC improvement of about three percent and a takeoff thrust increase of up to 8.8 percent can be accomplished through airflow changes which would require an inlet change, but not a frame size change. All modifications suggested appear attainable. Takeoff thrust increases up to 42 percent are possible if the frame size can be altered.

The P&WA engine could be modified to improve the cruise SFC by about four percent without changing inlet size. Attainment of the compressor and fan performance required to obtain this improvement will be difficult. The takeoff and transonic thrust increase of 12.5 percent will require an inlet size increase. This thrust increase requires a 10 percent increase in airflow which may be difficult to obtain in the present compressor size.

A further refinement by P&WA, raising the bypass ratio, looks promising. Takeoff thrust increases as high as 25 percent are possible without adding a turbine stage.

Performance gained from:	Engine changes required
	General Electric
Evolution of GE4/J5P cycle a. Increased airflow Step 1 - + 2.5% at T/O & +4.5% at cruise Step 2 - + 2.5% at T/O & +9.0% at cruise Step 3 - +13.0% at T/O & +9.0% at cruit * Step 4 - +35 % at T/O	a. Compressor redesign and resized inlet 1 - Overspeed, stator reset, and mater 2 - Overspeed, stator reset, and mater 3 - Compressor flaring, overspeed, sta substitution, enlarged augmentor, n 4 - Zero-staged compressor with redes
b. Maximum increased turbine inlet temperature ΔTIT = +100°F	b. Increased cooling flow and material sub enlarged turbine diaphragm
c. Maximum increased augmentor temperature ΔT A/B = +360°F	c. Modify fuel system to handle increased
	Pratt and Whitney Aircra
1-Component development a. Increased airflow at takeoff and transonic conditions	a. Reduced fan and compressor blade soli
b. Increased compressor efficiency (+6%), distortion tolerance, and surge margin	b. Slotted rotors and stators, variable ca reduced end-wall losses
c. Increased turbine inlet temperature, ΔTIT= +75°F	c. Material substitution in turbine blades
	d. Nozzle redesign for better utilization o
d. Improved nozzle performance ΔC_V= +.3%	e. Reduced blade solidity, controlled vort (lower rpm for same flow), short main nozzle weight through light weight mat in reverser design.
e. Reduced engine weight, Δ specific wt. - 10%	
2-Bypass ratio increase Increased airflow	a. Redesigned fan rotor, duct heater, nor ejector; resized inlet.

1

uired	Credibility
c	
inlet material substitution material substitution ɔd, stator reset, material ·tor, modified nozzle redesigned front 4 stages ial substitution with an eased flow	a. 1 - Airflow capability demonstrated on 475 PPS compressor 2 - Airflow capability demonstrated 3 - Requires a major redesign but is achieveable within present design. 4 - Requires a major redesign of the entire propulsion pod but is attainable through development of present engine design. b. Achievable through experience and attainable within present design. c. Achievable through experience and attainable within present design.
Aircraft	
ls solidity ɔle camber IGV, lades and discs tion of the pressure d vortex flow in turbine main burner, reduced t material substitution	a. Requires a redesigned fan/compressor. Short chord blades with higher stage loading will probably require material substitution on the basis of strength. b. Requires a major redesign. Efficiency increase is optimistic. c. Achievable through experience and attainable within present design. d. Attainable through development of present design. e. Experience will dictate amount of weight reduction attainable with part life and TBO used as limiting factors.
r, nozzle, reverser, and	a. Present turbine limitations suggest a supercharge of the primary compressor may have to accompany this.

2

Table 4-B Engine Growth Performance Comparisons

	GE	P&WA
1. Growth within engine frame		
A. 3 year growth	Step 1	Component Development
o Δ SLS max aug F_N	+ 5.8%	+ 7.0%
• Δ Transonic max aug F_N	+10.6%	+ 7.0%
• Δ Cruise SFC (constant F_N)	- 1.9%	- 2.2%
B. 5 year growth	Step 2	Component Development
• Δ SLS max aug F_N	+ 8.8%	+12.5%
• Δ Transonic max aug F_N	+16.0%	+12.5%
• Δ Cruise SFC (constant F_N)	- 2.8%	- 4.2%
2. Growth with enlarged frame	Step 3 (Compressor Flared)	Bypass Ratio (BPR=1.6)
5 year growth		
• Δ SLS max aug F_N	+20.9%	+25.5%
• Δ Transonic max aug F_N	+25.9%	+30.5%
• Δ Cruise SFC (constant F_N)	- 2.8%	- 6.2%
	Step 3 (Zero-Staged)	Bypass Ratio (BPR=2.0)*
• Δ SLS max aug F_N	+42.7%	+42.5%
• Δ Transonic max aug F_N	+48.6%	+45.5%
• Δ Cruise SFC (constant F_N)	(not given)	- 5.7%

* May require added turbine stage

5.0 INLET/ENGINE COMPATIBILITY

Complete inlet/engine compatibility can be developed for the B-2707 airplane using either the GE4/J5P or the JTF17A-21B engines. The inlet incorporates features which provide wide stability margins for engine generated disturbances, low circumferential distortion, and the means to adjust the moderate radial distortion to favor the particular engine selected. These features are discussed in detail in the Propulsion Report - Part A, Document V2-B2707-12, Sec. 3.0, and they include:

a. The throat bleed vortex valve which provides a seven percent flow stability margin when the inlet is operating at one percent supercritical.

b. The centerbody cone bleed scoop which, in conjunction with the vortex valve, provides a buzz stability margin varying from 68 percent at Mach 1.4 to nine percent at Mach 2.7.

c. Vortex generators on both the cowl and centerbody, which can be modified in effectiveness to vary the pressure patterns between the hub and tip regions of the compressor or fan inlet.

Of the two engines considered, the compatibility development would be the least difficult with the GE turbojet. This is due to the basic cycle, design, and control concepts involved.

Compatibility between the inlet and engine involves four basic considerations:

(1) Inlet distortion and inlet flow stability effects on the engine

(2) Engine flow stability effects on the inlet

(3) Control interactions and responses

(4) Inlet/engine flow matching

5.1 ENGINE INLET DISTORTION AND FLOW STABILITY EFFECTS

The measured overall compressor face distortion levels for normal inlet operation are shown in Fig. 5-1. Two operating limits for the engines are also shown. The lower limit (labeled steady state) is the maximum estimated distortion that can be accepted without any performance loss or degradation of design life. The upper limit (labeled transient) is the maximum estimated distortion that can be tolerated without engine surge, flame-out or excessive blade stress. Between these limits, some performance loss can be expected. The B-2707 inlet is designed to keep distortion below the steady state limit for all subsonic and supersonic cruise conditions, and to keep distortion below the transient limit during all aircraft and engine power transients including takeoff and climb accelerations.

The distortion levels, in terms of the General Electric distortion index (NDI), are below the specified limits for all conditions. This index is a term which weights the extent and distribution of the low total pressure regions at the compressor face, and incorporates a recognition of preferred radial distribution and the relative insensitivity of the compressor to small isolated regions of low total pressure. Figure 5-2 shows that even with extreme super-critical operation of the inlet, the Number Distortion Index (NDI) remains well below the limits. In the event of an inlet hydraulic system failure on takeoff, with the inlet centerbody full expanded, the NDI is eight percent below the level required to induce stall.

In terms of the P&WA distortion index, $\frac{P_{Tmax} - P_{Tmin}}{P_{Tave}}$ the distortion levels shown in Fig. 5-1 exceed the limits for no performance loss during a portion of the climb condition. However, the levels do not exceed the stall free operating limits for the engine. Figure 5-2 shows the margin for supercritical operation. While this margin is somewhat less than that for the GE engine, it is sufficient to allow for shock excursions resulting from power and inlet control transients.

Aside from the differences in distortion indices and limits employed by GE and P&WA, the

JTF17A-21B is considered to be inherently more sensitive to inlet distortion. The overhung fan rotor, short bearing span design requires high stage loading, short chord blading and no inlet guide vanes. Further, the primary gas generator flow has a relatively steep hub to tip inlet total pressure gradient, since it receives only the inner annulus of the inlet flow.

The GE4/J5P design employs a relatively moderate front stage loading, moderate aspect ratio blading and variable inlet guide vanes and stators.

Tests and studies of the inlet, inlet control and inlet/engine dynamics have revealed only one significant effect on the engine, and that is stall induced by the inlet unstart transient. In the event of an inadvertent unstart at high supersonic Mach numbers, it is anticipated that the sudden reduction in inlet pressure will cause a momentary engine stall.

Based on the J93 engine flight test data, the GE4/J5P engine can normally be expected to recover from the stall in a fraction of a second after unstart. Similarly, if the afterburner flames out, it will auto-ignite in a fraction of a second after the unstart. Flight test data on current supersonic aircraft indicate there is little likelihood of a primary burner flameout in either the GE or P&WA engine.

The JTF17A-21B is also expected to recover from the unstart induced stall in a fraction of a second. However, the duct burner fuel is automatically shut off, and the pilot must recycle the power lever through the duct heater ignition zone to affect a re-light.

5.2 ENGINE FLOW STABILITY EFFECTS ON INLET

The inlet control system is designed to respond to all but the most extreme engine induced flow variations (e.g., engine stall). If a reduction in flow exceeds a rate of 35 percent per second and an amplitude of 1/2 percent, the inlet normal shock will momentarily move into the vortex valve throat bleed slot. If the inlet is unstarted, the centerbody cone bleed scoop will prevent engine flow instabilities from inducing inlet buzz.

The GE4/J5P turbojet, with direct control of rotor speed, produces relatively slow rates of change. The inertia of the rotor, and the choking of the turbine diaphragm, dampens engine generated disturbances before they reach the inlet, permitting the inlet control to follow closely. Figure 5-3 shows the results of a mathematical model simulation study of an afterburner light-off at Mach 2.2.

The P&WA JTF17A-21B turbofan, with duct Mach number control of airflow and unchoked flow across the fan stage, will occasionally create flow variation rates exceeding the capability of the inlet control system. This will occur primarily during power changes. In these instances, the vortex valve will come into operation to hold the shock until the inlet control responds. Figure 5-3 shows the results of a mathematic model simulation study of a duct burner light-off at Mach 2.2.

5.3 CONTROL INTERACTIONS AND RESPONSES

Inlet/engine mathematic simulation studies to date have indicated acceptable control system interactions and responses.

When sufficiently refined simulation models are available, a study will be conducted to determine if the inlet control and the turbofan duct Mach number control interact across the low pressure ratio fan. Similarly, tests are required to determine if inlet distortion will carry through the fan and result in erroneous engine duct Mach number control signals.

5.4 INLET/ENGINE FLOW MATCH

Both engines provide the means to adjust engine airflow to match the inlet capture flow at the design cruise condition. This adjustment provides compensation for manufacturing and control tolerances.

The GE4/J5G provides for automatic airflow trim (engine rotor speed and secondary airflow control bias) to compensate for non-standard day cruise operation. This trim adjustment results in maximum propulsion pod thrust minus drag for any ambient temperature.

The P&WA JTF17A-21B airflow control schedule maintains good flow matching, with maximum thrust minus drag, for ambient temperatures at and above standard. There is no trim capability for cold day operation with the Boeing selected engine flow schedule. As cruise temperatures decrease from standard, the nor-

TRANSIENT – TIME LIMITED
NO STALL OR SURGE
NO ENGINE FLAMEOUT
MAY BE SOME
PERFORMANCE LOSS

STEADY STATE – NO PERFORMANCE LOSS
NO CHANGE IN ENGINE
LIFE

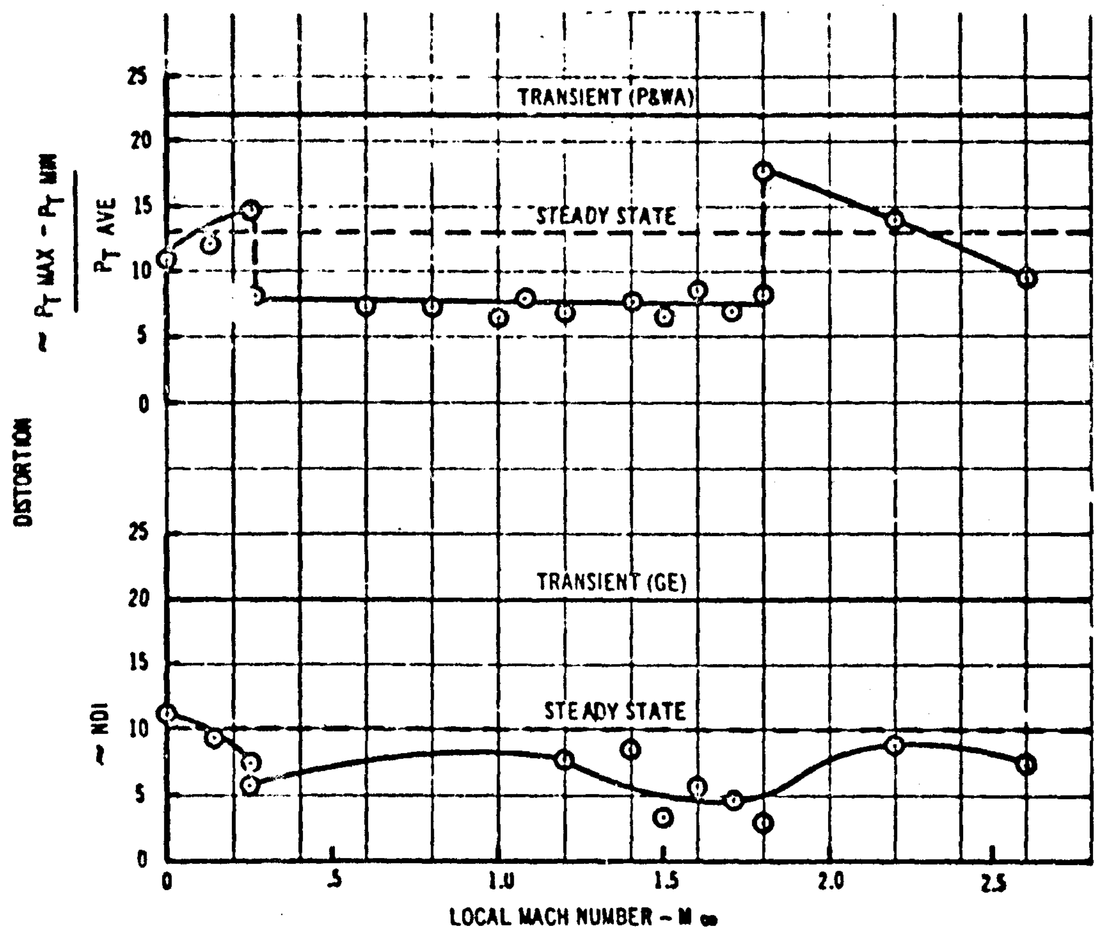


Figure 5-1. Summary of Inlet Distortion

mal shock becomes progressively more supercritical (more stable) with some resultant loss in inlet recovery.

The inlet bypass system is sized for the descent condition with the windmill brake applied. The JTF17A-21B engine passes a larger mass flow ratio under these conditions than does the GE4/J5P. As a consequence, the bypass system is approximately 25 percent smaller for the turbofan. During subsonic cruise operation, the turbofan again has a mass flow ratio advantage; this is equivalent to a two percent SFC subsonic advantage.

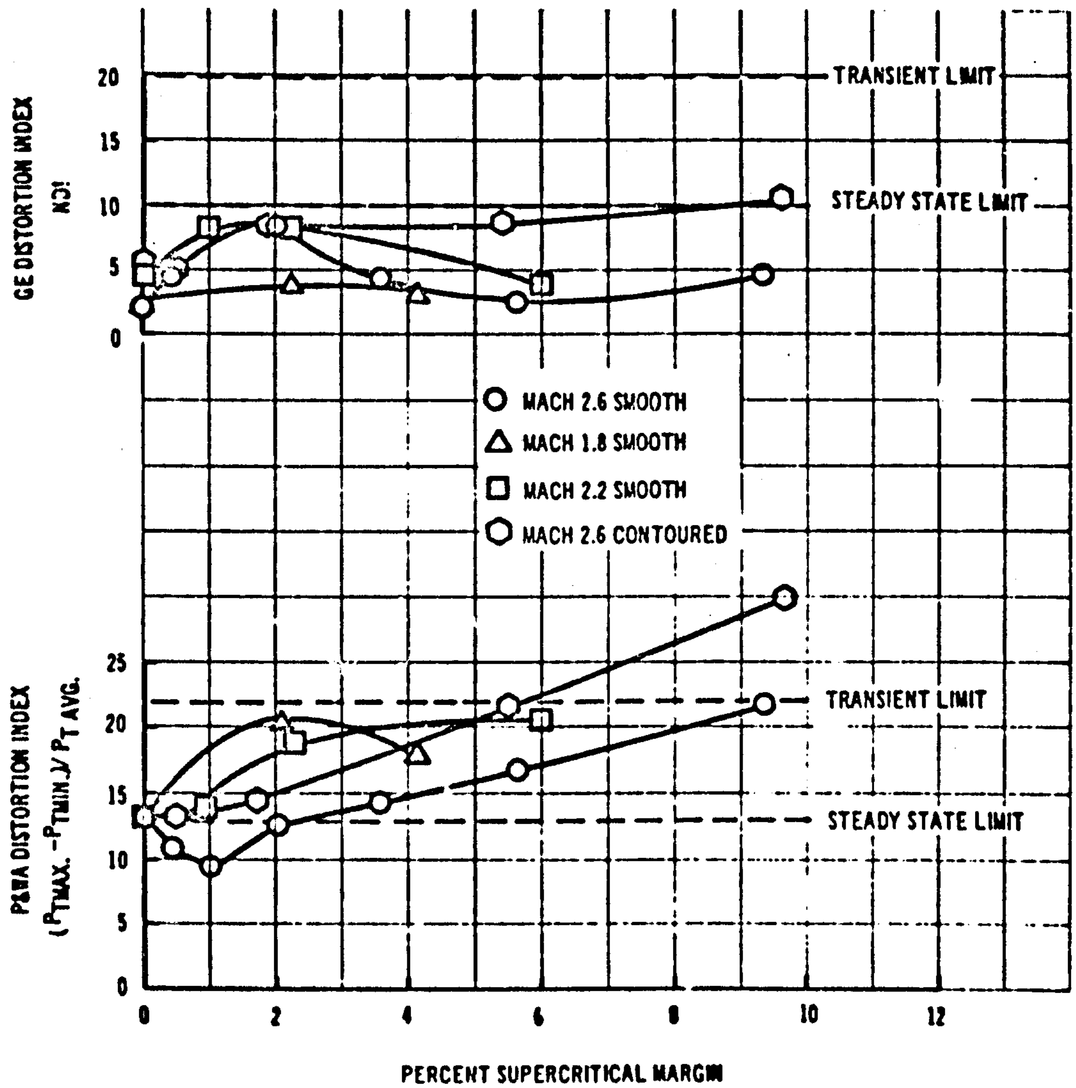


Figure 5-2. Inlet Distortion Versus Supercritical Margin

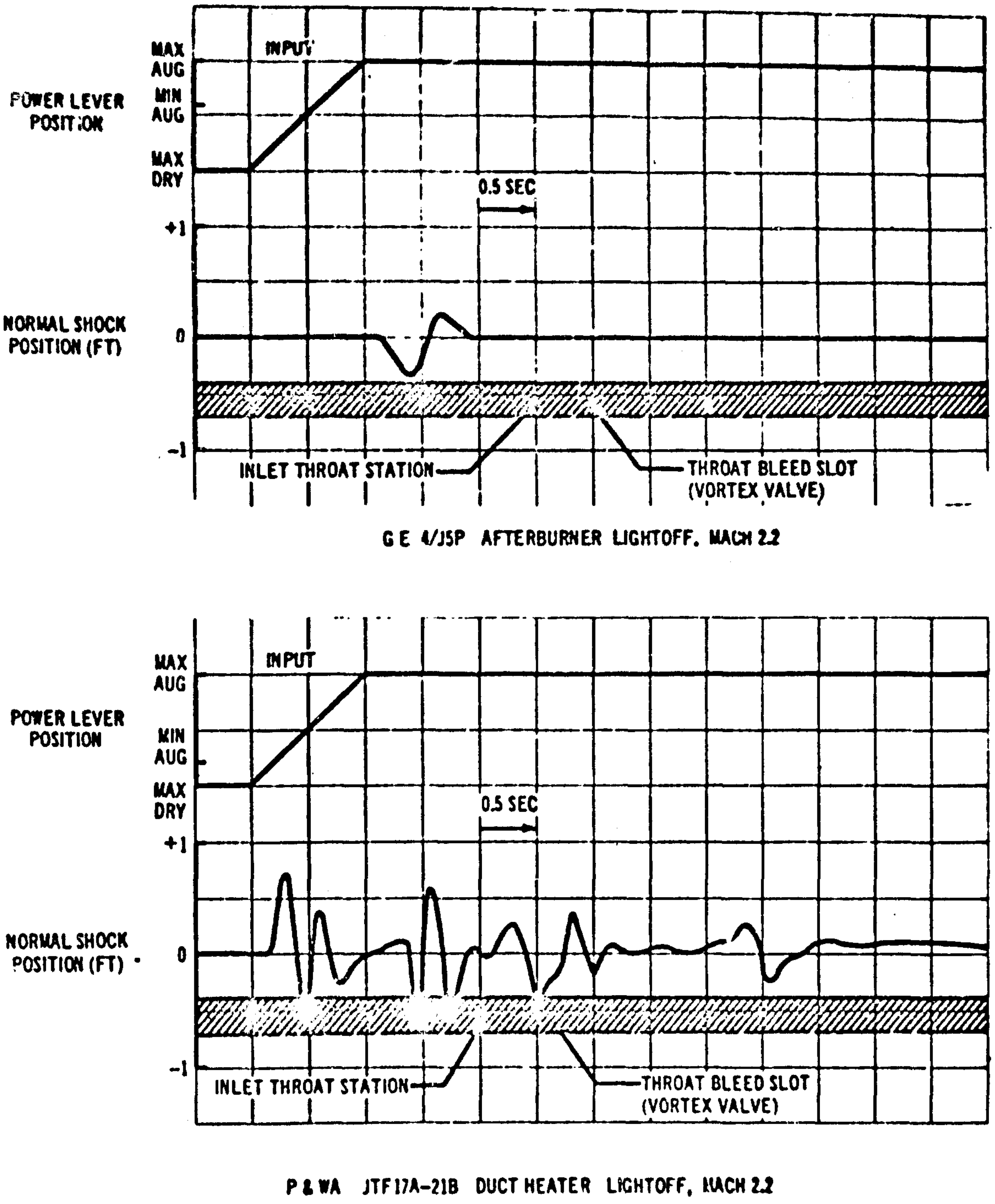


Figure 5-3. Augmentor Lightoff Simulation Results

6.0 ENGINE DEVELOPMENT RISK

This evaluation is based on information received from the two engine manufacturers during the SST program. The most specific and recent information was obtained during a visit by Boeing representatives to the manufacturers' facilities in mid-July, 1966, and from preliminary draft copies that represented portions of the Phase II C engine proposal documents.

In general, the propulsion system and components for the SST represent an advance in technology over present flight propulsion systems. Because of this, each component of the engine represents a development risk, to some degree, and each component considered was placed in one of three risk categories. Briefly, component/system performance, life, complexity, and weight were considered in placing that item in a risk category. The judgement factors used were:

- Design Goals
- Demonstrated Performance
- Past Experience
- Technical Capability
- Design Approach

In some instances component design and performance were, when measured against today's demonstrated technology, of such a low risk as to not warrant being placed in one of the three major risk categories (i.e., a normal development program should ensure specified performance).

The risk categories are:

a. Category 1 - The component or design has some questionable aspects at this time. Some problems are foreseen, and an above average success, in a well run development program, will be required to accomplish the design goals. The performance goals will probably be reached.

The SST Program implications are:

Increased development program costs
Reduced parts life
More complexity
Minor program delays

b. Category 2 - A component placed in Category 2 suffers the same risk as Category 1, but to a higher degree. Moreover, additional program implications are present, especially if the goals are not reached. Quite possibly, not all goals will be reached, particularly in early commercial service.

The SST Program implications (additional to Category 1) are:

Payload-range decrement
Increased direct operating cost (DOC)
Increased program delays

c. Category 3 - A Category 3 item is one which is a risk item as in (1) and (2) above, but has the potential of significantly affecting the overall program. The attainment of specified goals and performance is doubtful.

The SST Program implications (additonal to Category 1 and 2) are:

Major program delay
Major program redirection

Note: The material in the following sections is based on a more complete treatment of the subject contained in Appendix A.

6.1 GENERAL ELECTRIC GE4/J5P ENGINE

6.1.1 Compressor

The 620 lb/sec nine stage compressor has a sea level static (SLS) pressure ratio of 12.3 at 86.0 percent efficiency. Cruise efficiency will be 84.5 percent. The demonstration engine has a 475 lb/sec eight stage compressor, which has been rig tested. The demonstrated test results

and lack of known mechanical design problems indicate that the compressor performance closely approximates the design goals. The distortion test results reported to date on the eight-stage demonstrator compressor indicate that distortion problems will likely be minimized. In addition, the inherent flexibility of a variable geometry design allows significant changes in compressor performance without redesign of the compressor, should problems arise. Overall, the GE compressor is not considered to be in one of the major development risk categories.

6.1.2 Main Burner
The annular main burner has a film cooled wall, giving wall temperature of 1,500°F. At cruise the burner efficiency is 98.75 percent, with a 7.2 percent total pressure drop.

The design space heat release rate and the exit temperature profile are two aspects requiring further development work. At this time a full scale air flow rig has not been used. General Electric's past experience with similar burners, and the test program and demonstrated performance to date are encouraging. Overall the GE4/J5P main burner does not fall into one of the major risk categories.

6.1.3 Turbine
The turbine is designed for a gas temperature of 2,250° F inlet temperature at takeoff, in climb and in acceleration, and for 2,200°F in cruise. Cruise efficiency is 90.3 percent.

The GE4/J5P turbine work output requirement is such that the two stage turbine easily meets this work requirement. General Electric has other operational engines such as the J79 and J93 running at these turbine work and efficiency levels. The turbine is therefore a conservative aerodynamic design. The design offers the flexibility to extract more work if this should be required during engine development. The aerodynamic performance of the GE turbine is not considered a risk item.

Mechanically, the first stage turbine blades operate at a tensile stress of 13,300 psi at the critical section and have an average metal temperature of 1,550°F. Total cooling flow is 12.3 percent. The blades are made of cast Rene'69, the first stage vanes of cast X-40, and the other vanes of Rene' 77. There is a possibility that GE may have design problems with film cooling development and achievement of the desired material properties in Rene' 69, or coating problems if Rene' 100 is used as a substitue. The GE4/J5P turbine blade life is therefore classified as a Category 1 risk item.

6.1.4 Afterburner
The maximum afterburner (A/B) gas temperature is 2,840°F. Afterburner entry temperature in cruise is 1,600°F. Augmentor temperature rise in cruise is about 300°F, and afterburner chemical combustion efficiency is about 99 percent.

The GE augmentor is based on actual flight experience with the J79 and J93 engines. An important question is one of commercial life of augmentor parts. General Electric has set a design life goal of 4,000 hours without repair. Attainment of this goal will be difficult. Because of the parts life question at this time and the consequent effects on airplane dispatchability, the augmentor life is classified as a Category 2 risk item.

6.1.5 Nozzle/Reverser
The lack of substantiating test data from models exactly duplicating the present two-stage ejector nozzle design, the apparent inexperience of GE with this particular ejector concept, and the strong effects of nozzle performance on payload-range, make the nozzle a Category 2 development risk item.

The thrust reverser should perform as quoted by GE, but the life and reliability factors make this component a Category 1 risk item.

6.1.6 Engine Weight
Boeing does not feel qualified to present a detailed analysis of the weight of the GE4 engine. There exists no operational precedent for the nozzle-thrust reverser system as offered. Because of this, there is every reason to consider the possibility of weight increase above the present engine company estimates. This factor, together with the consequences of overweight on airplane performance, classify the weight of the GE engine as a Category 2 development risk.

6.1.7 Controls and Engine Dynamics
General Electric is offering a control system

which is virtually identical to previous operational control systems. The compressor variable stators offer a degree of flexibility in engine-inlet compatibility problems.

General Electric has an experience background that is directly applicable to control system simulation and has shown that this simulation gives realistic predictions.

Afterburner light-off does not present problems for the inlet even with a delayed light-off. The afterburner operates during most of the mission under auto-ignition conditions. A computer run with forced blow-out showed engine airflow transients within Boeing limits.

The GE4/J5P engine will probably experience a momentary compressor stall due to an inlet unstart. The main burner will remain lit at all times. The afterburner will probably blow out but will auto-ignite within a fraction of a second.

General Electric has initiated a well defined program on compressor surge margin and distortion tolerance. Distortion test data for the eight stage compressor looks good for this phase of the program. However, the nine stage compressor differs significantly from the eight stage demonstrator (aspect ratio of first stage blades of 2.4 and 1.3 respectively).

Thus the distortion tolerance of the nine stage compressor remains to be demonstrated. In case of persistent distortion or dynamic control problems GE can incorporate the variable stators in the high response part of the control system.

For these reasons, the controls and dynamics of the GE4/J5P engine constitute a Category 1 development risk.

6.2 PRATT & WHITNEY JTF17A-21B ENGINE

6.2.1 Fan and Compressor

At sea level static design conditions the fan has a tip speed of 1694 fps, a tip relative Mach number of 1.67, and efficiencies of 78.8 percent for the outer annulus, or duct side, and 88.8 percent for the inner annulus, or engine side. At cruise the efficiencies are 80.8 and 89.8 percent, respectively.

The fan is still in an early development state and performance goals have not been demonstrated. The lack of distortion testing together with the fluid state of the fan design, make the achievement of stated fan performance goals a Category 1 development risk item.

At sea level static the six-stage HP compressor has a design pressure ratio of 4.84 with 85.9 percent efficiency. At cruise the efficiency is 86.8 percent. The HP compressor design is still in a early development state, performance goals have not been demonstrated and distortion test results are not available. These factors indicate that the HP compressor is a Category 1 development risk. Considering that the HP compressor must be developed to accept fan hub flow, the overall compressor section of the JTF17A-21B is viewed as a Category 2 development risk.

6.2.2 Main Burner

The main burner is an annular ram induction burner with a burner efficiency of 99 percent. The burner liner peak temperature is 1,800°F to 1,850°F.

The ram induction concept is new, and no flight experience is available as a technological base. However, test performance to date is encouraging and there is reason to believe that P&WA can achieve their goals. The main burner therefore does not fall into one of the risk categories.

6.2.3 Turbine

The turbine is designed for 2,300°F turbine gas inlet temperature at takeoff and acceleration, and 2,200°F in cruise. The cruise efficiencies for HP and LP turbine are 86.9 and 88.0 percent.

The JTF17A-21B turbine is a highly loaded aerodynamic design with high cascade Mach numbers and low exit hub/tip ratios. It appears that the efficiency will be difficult to achieve as the specified rotor tip clearances of 0.02 - 0.035 inches will be difficult to maintain in operational engines. An increase to a more conventional 0.08-inch clearance will cost about 2 percent in turbine efficiency. The JTF17A-21B turbine aerodynamic performance is classified as a Category 2 development risk item.

Typically, a fan engine turbine blade has rather high stresses at mid span. The average metal temperature of 1,640°F together with the convection type of cooling, and the material selected

for the turbine blades (PWA 658), make it difficult to obtain the stated creep life of 10,000 hours. The JTF17A-21B turbine life is classified as a Category 1 development risk item.

6.2.4 Augmentor
The augmentor, a ram induction burner, is designed for a maximum gas temperature of 3,140°F and a maximum temperature rise of 2,800°F. The nominal cruise temperature rise is around 930°F at a chemical combustion efficiency approaching 100 percent. Film cooling of the liner results in metal temperatures of 1,200°F to 1,400°F in cruise. Light-off occurs at a fuel to air ratio of 0.002.

The burner concept is new, and is without past supersonic flight experience as a technology base. The augmentor must perform under a wide range of flow conditions from the fan; and at high combustion efficiency levels. Because of the effect of duct burner efficiency on cruise SFC the duct burner is classified as a Category 2 development risk item. These comments are based on steady flow operation. Duct burner dynamic considerations are treated later.

6.2.5 Nozzle/Reverser
Although P&WA tests tend to substantiate the nozzle performance level at supersonic cruise, further development is required to achieve the nozzle performance goals at other flight conditions. In view of the strong affects of nozzle performance on airplane performance, the nozzle is considered a Category 2 development risk item until specific tests prove that the performance levels can be attained.

Pratt and Whitney Aircraft reverser model tests indicate that the reverse thrust design goals will be met. However, control of reverse gas flow direction and distribution is expected to present problems with the present design, in which the flow must exit through the blow-in-doors. The flow angle is such that the flow could attach to the engine nacelle and enter the engine inlet. For this reason, the reverser concept is considered to be a Category 2 risk item.

6.2.6 Engine Weight
Boeing does not feel qualified to make a detailed analysis of the weight of the JTF17A-21B engine. There exists no precedent for the offered nozzle-thrust reverser system and targeting requirements to avoid ingestion and impingement on aircraft surfaces, may lead to a weight increase. There is reason to expect a weight increase in the engine fan/compressor as a part of the effort to achieve the required steady state distortion and dynamic distortion tolerance.

These factors classify the weight of the JTF 17A-21B as a Category 2 development risk item.

6.2.7 Controls and Engine Dynamics
Based on the analysis available at this time, there are several areas of concern relative to the P&WA control concept.

In the duct heating turbofan cycle, the fan inlet airflow responds very quickly to downstream duct pressure conditions because pressure transients downstream of the fan can travel upstream, through the fan. For this reason, transients due to duct heater lightoff, duct heater blowout, changes in augmentation level, and normal fuel flow transients, can cause fan airflow variations. In the P&WA engine control system these airflow transients tend to be damped during duct burning by the variable duct nozzle, which is controlled by measurement of duct airflow. This flow is measured by a pitot tube located downstream of the fan and upstream of the duct burner. The purpose of the measurement is to keep the total fan inlet flow constant at any power setting above max dry. Several aspects of this control concept are of concern:

a. The pitot tube must sense total to static pressure differences accurately in a stream flowing at around Mach 0.5. A reliable and true indication of airflow by this technique is a difficult instrumentation problem. This is mainly due to the effect of steady state and/or dynamic changes in the radial or circumferential pressure profiles at the pitot tube (in contrast to pressure changes without profile shifts). Such changes could originate upstream of the fan, within the fan itself due to changes in operating conditions, or from internal duct pressure disturbances.

b. Because changes in fan inlet flow will occur as the direct result of the duct transients described above, the HP compressor stall margin provided in design, under both distorted and fluctuating flow, must be adequate. However, there are no provisions in this engine design to

balance the load between the fan and compressor in the hub flow region (such as variable stators and variable primary nozzle). P&WA data show that a duct burner blowout will cause fan stall, mainly because the duct pitot tube and nozzle area control cannot respond fast enough. There are no data taken either on the fan or compressor rigs to date which show that this compressor or fan design can tolerate the resultant disturbances or distortion.

c. In the event of sufficiently large duct pressure transients causing more than a four percent increase in indicated duct airflow, the present control automatically shuts off the duct fuel flow. An airflow transient greater than four percent will result in a duct heater blowout, which requires a shutdown of heater fuel flow. This fuel flow shutdown is essential because an uncontrolled heater relight is unacceptable. Only a controlled relight at minimum fuel/air ratio produces sufficiently small airflow transients. Relight at high fuel/air ratios would produce unacceptable airflow transients.

At the present time, the P&WA dynamic analysis procedures are not sufficiently complete to allow a study of the control of the ducted fan cycle in depth; and fan/compressor component tests to determine response to dynamic pressure disturbances have not yet been performed. Such work is planned by P&WA in the development of the JTF17A-21B engine.

Because of the unknowns regarding the control system concept, and because the response of the engine system to transient disturbances may, to a certain degree, be fundamental to the turbofan cycle, Boeing believes the control and engine dynamics represent a Category 3 development risk. This is believed to be true until sufficient component and/or engine testing, together with detailed mathematic model studies, have shown that no fundamental problems exist.

7.0 INSTALLATION DIFFERENCES

The installed performance of the engine in the airplane depends upon the physical position of the powerplant with respect to other components as well as the interaction of the engine and the airframe. The following discussion will be devoted to differences between the B-2707 (GE) and B-2707 (P&WA) with respect to safety, maintainability, life, and structure.

Performance comparisons will be made only when a significant effect upon installed performance is indicated due to the arrangement or operation of the engine. The intent is to present an overall comparison of the two engines as a functional part of the B-2707 airplane.

7.1 REVERSE THRUST

The reverse thrust performance is compared on the basis of normal as well as emergency performance margins. The first is related to airplane kinetic energy and brake life while the second can be evaluated primarily in terms of safety. The GE powered airplane is superior when normal landing conditions prevail due to the greater amount of reverse thrust available from the engine. The comparison of the amount of kinetic energy reacted by four engines is:

GE	P&WA
31.5×10^6 ft-lb	19.1×10^6 ft-lb

When a comparison is made based on a landing where the brakes have completely failed, it shows that the value of reverse thrust provided by the General Electric engine will reduce the landing roll by 800 ft more than the P&WA engine. This is based on a reverse thrust static difference between the engines of approximately 10,000 lbs (reverse thrust for the Pratt & Whitney Aircraft engine is 14,080 lb while reverse thrust for the General Electric is 23,200 lb. The General Electric-powered airplane rolls 5,500 ft under these emergency conditions, while the P&WA engine-powered airplane will roll 6,300 ft after touchdown. With the use of reverse thrust as well as the brakes for a normal reaction time from the pilot and average runway conditions, a difference in landing distance of 100 ft can be expected as shown in Fig. 7-1.

The efficiency of the installed reverser system is greatly influenced by the design of the reverser-stabilizer region where a portion of the reverse flow is directed forward over the top of the stabilizer.

Due to the more forward location of the General Electric engine, the position of the reverser doors on top of the stabilizer tends to be restrictive as to the portion of the periphery which can be opened to allow the use of a reverser. The P&WA reverser doors are located in a more rear location and therefore can be larger due to the increase in nozzle periphery area available. (See Fig. 7-2 for reverser door locations.) Although this accounts for only 3/11 of the total periphery, it is the flow in this region of the airplane which is the most effective in contributing to stopping. This is due to the fact that no other airplane surfaces or ground effect will be changing the airplane drag and hence the effective reverse thrust.

7.2 SIZE AND SHAPE

7.2.1 Dimensions and Weight

The General Electric engine is installed in a pod which is 446 in. long and weighs 14,312 lb, while the Pratt & Whitney Aircraft engine is installed in a pod which is 345 in. long and weighs 13,865 lb. The difference in the maximum diameter between these two pods is 1.0 in. (89 in. for the General Electric and 88 in. diameter for the Pratt & Whitney Aircraft engine).

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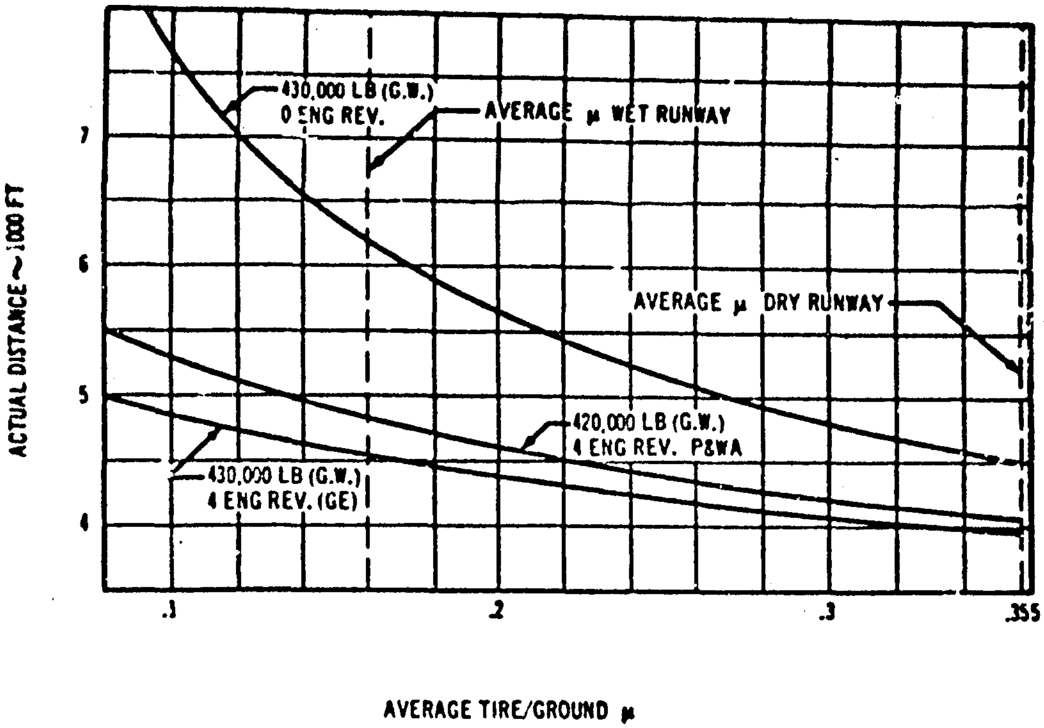

Figure 7-1. Landing Roll Distance

Table 7-A shows several significant weight differences. Part of the weight difference of the total installation is due to the engine supporting structure.

7.2.2 Horizontal Stabilizer Trough
The difference in pod length and the location of the reverser doors makes the trough in the horizontal stabilizer upper surface significantly different in the two installations. Due to the more forward location of the General Electric engine, it is necessary to increase the angles which are part of the fairing between the nacelle and trailing edge of the stabilizer. Figure 7-3 shows the angles in the trough region above the nacelle. It shows that the trough angles associated with the GE installation are steeper than those for the P&WA engine installation.

7.2.3 Loads

Due to the larger size of the GE powerplant pod, the aerodynamic loads are from 2 to 8 percent higher. The differences in shear loads and moments are shown in Table 7-B. The table also shows that the loads due to engine seizure are much greater in the case of the General Electric engine.

7.3 SAFETY

7.3.1 Inlet Protection
The probability of ingestion of foreign materials thrown by the landing wheels is the same for the two engines. The further aft location of the P&WA pods is a small advantage in avoidance of ingestion of hard objects, such as rocks, concrete fragments, etc., and a small disadvantage in avoidance of water and slush spray. These conditions vary with rolling speed and quantity of foreign material on the runway. The inboard and outboard flaps provide a large measure of inlet protection for both engine installations, and in this respect, no substantial difference between the two engine types exists.

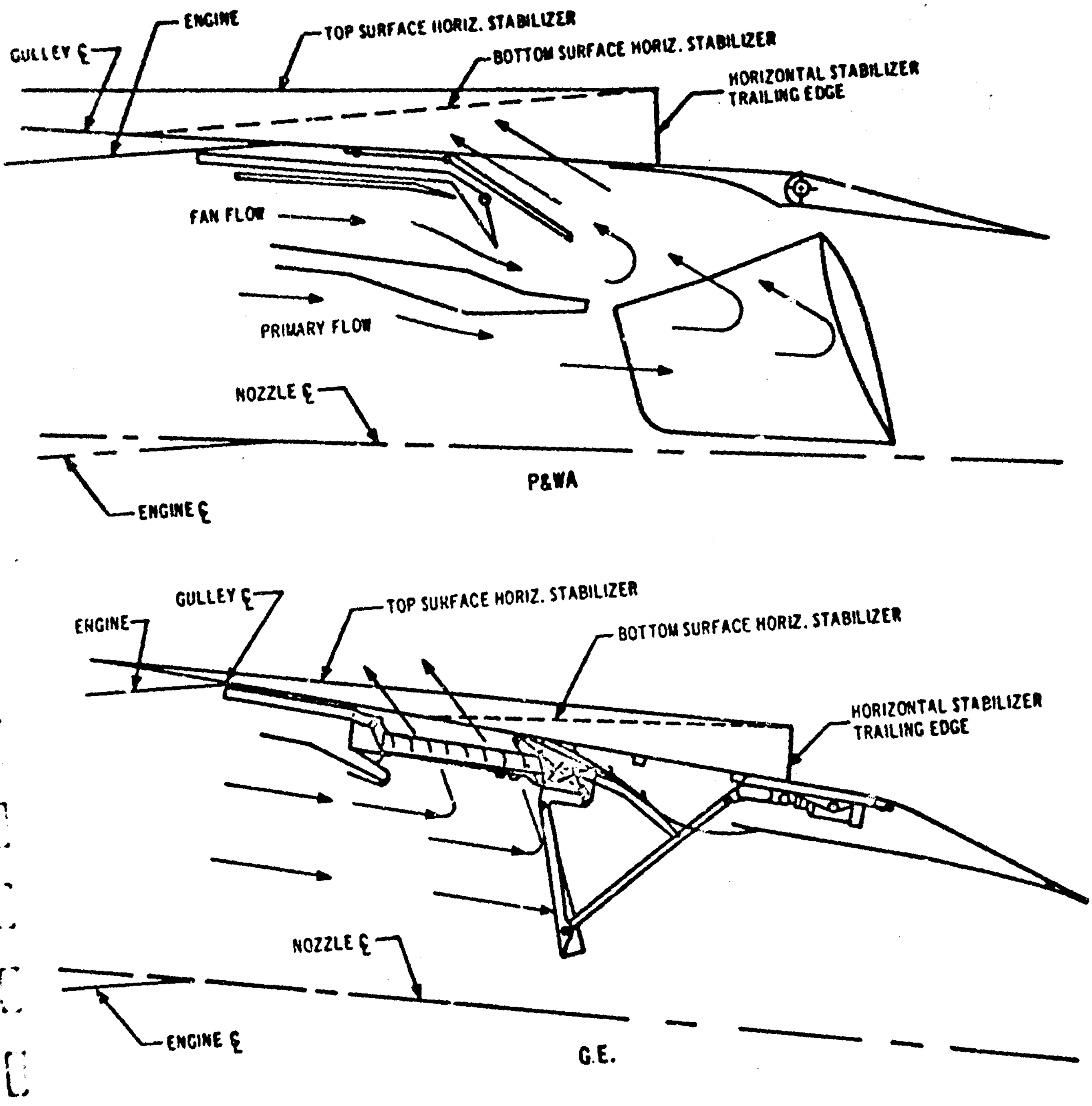

Figure 7-2. Reverser Door Location

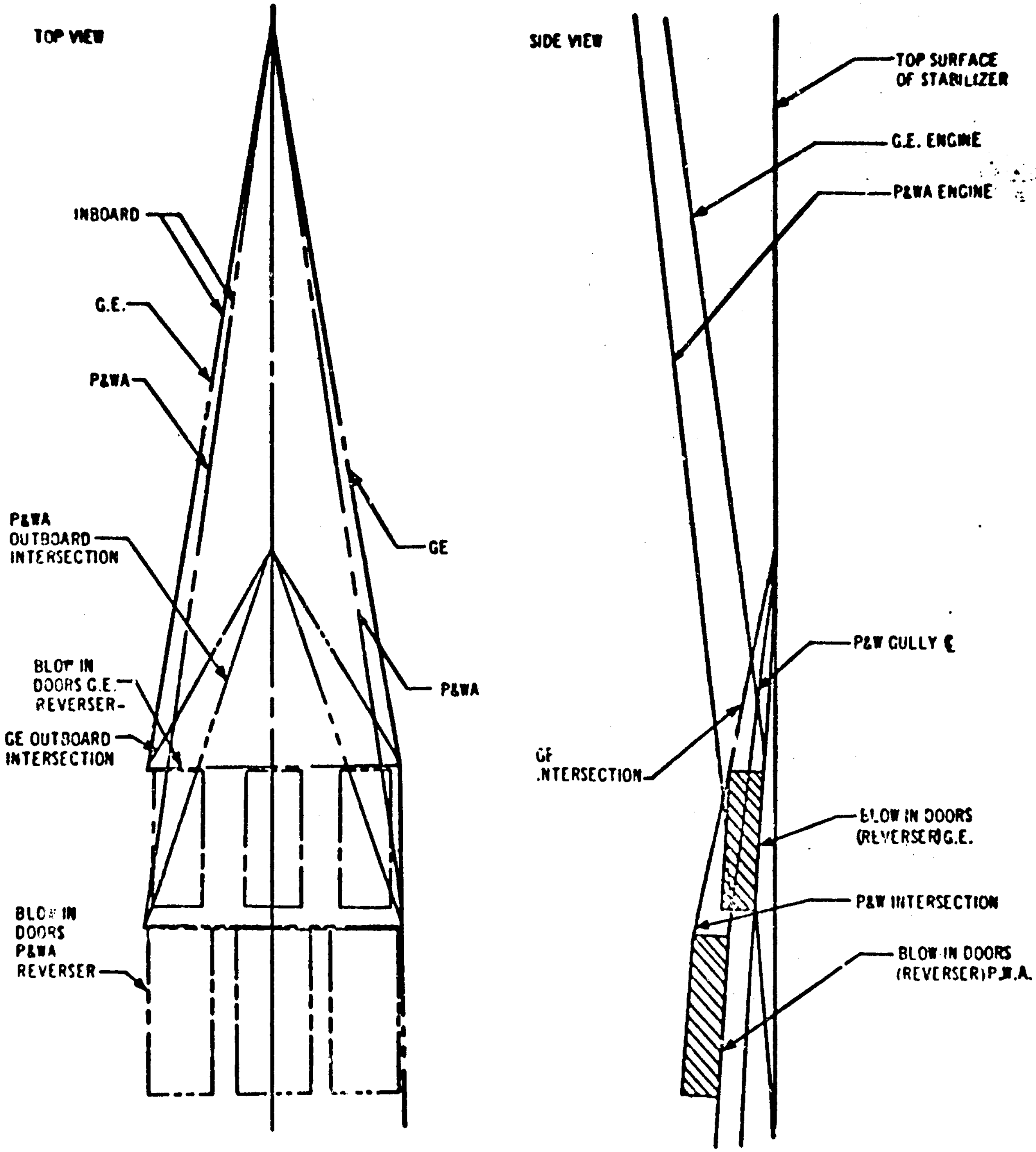

Figure 7-3. Gully Angles

Table 7-A Weights

	P&WA LBS		GE LBS	
Inlet		9,940		8,280
Diverter	200		180	
Cowl and door	4,160		3380	
Engine cowl panels		900		1,3[illegible]0
Aft engine cowl		-		600

Table 7-B Nacelle Loads

Resultant Load Acting at Engine Pod, C.G.	P&WA Magnitude (Max)	GE Magnitude (Max)
Aerodynamic sideload on entire pod	33,500 lb	34,500 lb
Moment due to aero. sideload on entire pod	3390 in-kips	3660 in kips
Aerodynamic vertical load on entire pod	22,300 lb	23,000 lb
Moment due to aero. vertical load on entire pod	2260 in-kips	2440 in-kips
Roll moment due to engine seizure	820 in-kips	3370 in kips

Engine mounting system loading

7.3.2 Ingestion
The General Electric engine has a better capability to ingest forms of ice which the airplane may encounter. The General Electric engine is guaranteed to continue to run after ingesting 3 in. diameter hailstones while the Pratt & Whitney engine has been guaranteed for 2 in. diameter hail. A similar difference exists when comparing the engine manufacturers' stated capability for ice slab ingestion. The GE engine can ingest 2 in. thick slabs while the P&WA can ingest only 1/2 in. thick slabs.

7.3.3 Windmill Power
The airplane has been designed to provide for control in case of the simultaneous failure of all four engines. To maintain this control a minimum horsepower requirement has been established which will provide for actuation of the aerodynamic control surfaces as well as emergency subsystems. With the Pratt & Whitney Aircraft engine, it is possible to extract 65 hp from the auxiliary drive system on each engine while it is windmilling. This amount of power is sufficient to provide the emergency power required to control the airplane.

Since the General Electric engine cannot deliver the required horsepower while windmilling, a retractable ram air turbine will be installed in the wheel well.

7.3.4 Rotating Machinery
With the engines pods mounted on the stabilizer, it is desirable to place the rotating machinery at a station location which is as far aft as possible and thus provide no interference with other airplane components in case of engine failure. The design criteria used for locating the engine are such that in the plane of the rotating machinery, there will be no primary system elements which

could be damaged in case of a failure of the turbine or compressor.

The location of the rotating machinery in the General Electric engine installation of the airplane is farther forward than in the Pratt & Whitney Aircraft engine installation. This forward location of the inboard engines limits the storage of fuel in the stabilizer in the region above the compressor of the GE engine as shown in Fig. 7-4. With the farther aft location of the compressor in the Pratt and Whitney Aircraft engine, it is possible to use additional portions of the stabilizer trailing edge for tankage and thus increase the fuel capacity by 5628 lbs.

7.3.5 Fire Protection

The possibility of an engine primary combustor burnthrough due to burner can failure is reduced in the P&WA engine compared to the GE engine, because the primary burner is surrounded by the low pressure engine fan ducting which would block and contain the burnthrough.

7.3.6 Mode Selector

The greater number of modes will require more attention from the flight crew of the B-2707 (GE) airplane. The crew attention factor is compensated by the fact that in the P&WA engine it is necessary for the flight crew to adjust the exhaust gas temperature. However, this adjustment will be made infrequently. It will also be more difficult to provide suitable linkage adjustments between the selector valve on the GE engine and the control on the flight deck. The GE selector valve requires the use of five positions and an angular travel of 132°, while the P&WA engine requires only three positions with a total travel of only 90°. The P&WA selector valve has more generous allowances for the deadband at each end of the travel. A comparison of the positions for the selector valves is shown below.

P&WA	GE
	Windmill Brake
Shutdown	Shutdown
Run	Descent
Secondary Air	Run
	Cruise & Holding

7.4 MAINTAINABILITY AND LIFE

7.4.1 Accessories

The difference between the two engines with respect to location of engine mounted accessories makes the accessories on the General Electric engine operate within an ambient temperature area 150°F above the temperature of the engine mounted accessories on the Pratt & Whitney Aircraft engine.

To partially offset the more severe environment, the accessories on the GE engine have been grouped into a capsule which has a double walled inner surface that is provided with a supply of cooling air. Additional cooling is attained from the fuel components in the compartment. Since the accessories on the Pratt & Whitney Aircraft engine are mounted in an exposed arrangement around the bypass air duct, the ambient temperature will be only 550°F. This makes it unnecessary to provide a separate cooling compartment although fuel cooling is used for those items which are especially temperature sensitive.

The location of the augmentor on the two engines does not significantly affect the temperature of the components mounted on the engine compressor case. The P&WA engine depends upon duct heating for thrust augmentation, while the GE engine depends upon the afterburner. Generally, only small quantities of augmentation are required during cruise flight and the duct heater effectively provides an additional layer of low pressure cooling air between the main burner and the outer engine case.

There is no significant difference between engines in terms of maintainability. This is due to the fact that the pod cowling provides the cover for the capsule on the General Electric engine. To gain access to either engine accessory area it will be necessary to open the cowl panels. The accessories in the compartment of the GE engine are packaged closer together and require additional time for servicing. The package concept will preclude the use of more than one mechanic at a time working on accessories. Distribution of accessories around the outside of the P&WA engine makes it possible for more than one

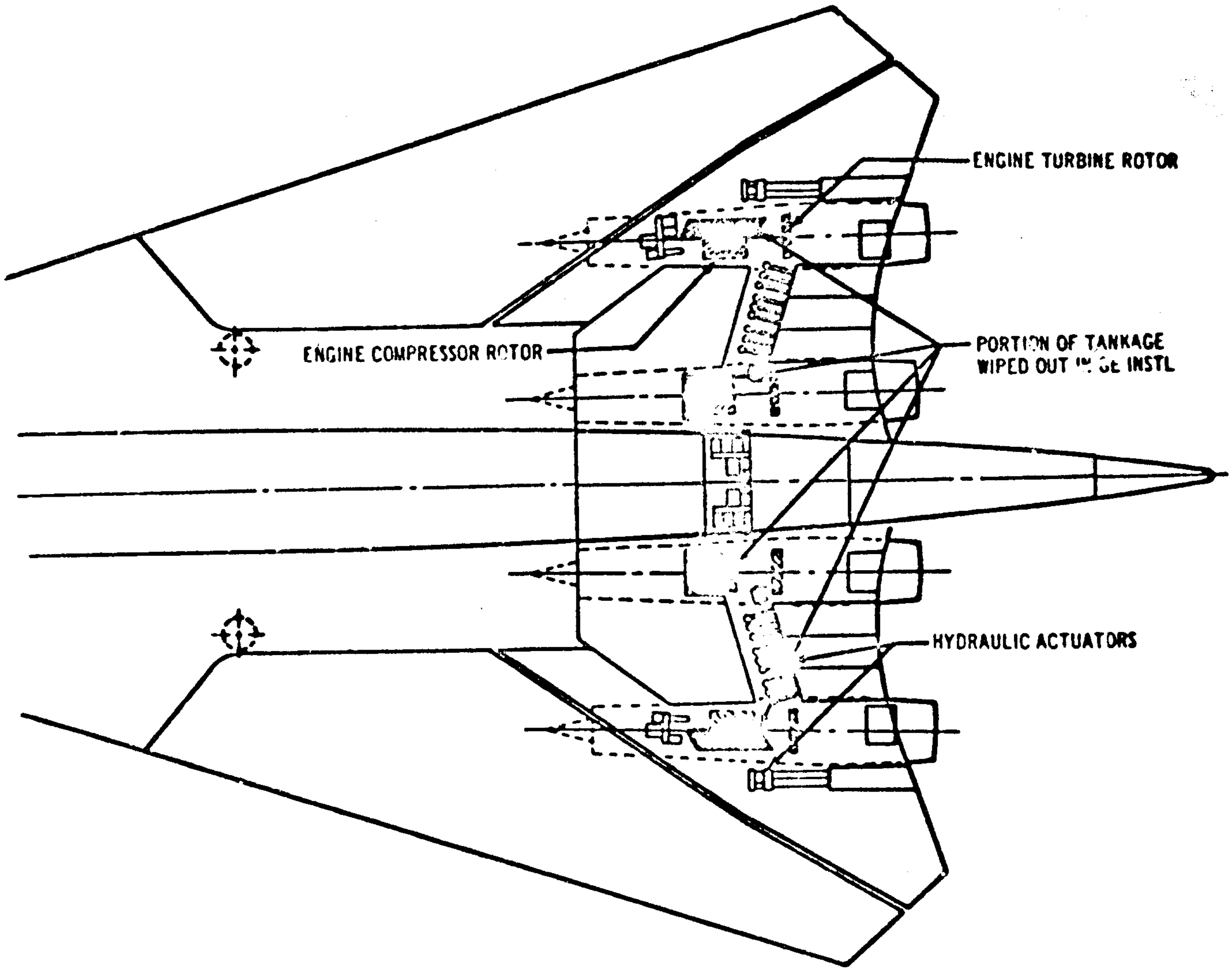

Figure 7-4. Fuel Tank Location

mechanic to work on engine accessories simultaneously.

7.4.2 Accessibility

The inspection of the P&WA engine interior through hatches will require more time because of the concentric duct around the primary compressor case. The GE engine will require the removal of only a single cover to reach the same equivalent point within the engine.

7.5 OPERATIONS

The number of controls for mode selection and thrust control are the same for both engines; however, it is necessary to provide a manual trim capability in the Pratt and Whitney Aircraft engine to compensate for exhaust temperature increases with engine usage. Since both engines have instrumentation which provides information on exhaust gas temperatures, there is no difference with regard to number of flight deck displays required.

8.0 MAINTAINABILITY, RELIABILITY, AND SAFETY

8.1 MAINTAINABILITY

Maintainability involves the evaluation of maintenance effort and maintenance time. Maintenance effort is the product of maintenance frequency, which is primarily a function of reliability, and maintenance manhours which is a product of the number of men required and the time required to accomplish a maintenance task. This evaluation directs itself to comparing the expected task time and number of men required to maintain the respective engines by review of the designs for maintenance features and comparing stated objectives by both manufacturers. This review does not emphasize the frequency of maintenance. The frequency of maintenance must be derived by the reliability evaluation.

8.1.1 Engine Comparisons

A comparison of GE4/J5P and JTF17A-21B engines is presented in Table 8-A.

Table 8-A. Maintainability Features

Feature	P&W JTF17A-21B	GE4/J5P
(1) Maintenance frequency		
(a) Design life objectives with repair		
Major Cases	50,000 hours	unlimited with repair
Discs and Rotors	20,000 hours	36,000 hours
Easily replaceable parts	10,000 hours	
Compressor blades		18,000 hours min.
Turbine Blades		12,000 hours min.
Sheet metal and wear surfaces		12,000 hours min.
(b) Unscheduled maintenance rate	.200/1000 E.H.	.250/1000 E.H.
(c) Inspection frequency Hot Section Inspection	5000 hours Also on-airplane borescope provided	No time stated. On-airplane with borescope
(d) TBO Goals	5000 hours in 4 years with on-condition objective	No hours stated. On-condition objective
(2) Maintenance Manhour Goals		
Unscheduled Line	None stated	31.7 MMH/1000 E.H.
Scheduled Line	None stated	113.3 MMH/1000 E.H.
Repair and Overhaul	None stated	500.0 MMH/1000 E.H.
Engine Sections (Typical) Remove and Replace		
Exhaust Duct/Case	9.5 Elapsed hours 18.0 Manhours	1.9 Elapsed hours 4.9 Manhours
Front Section Assy (Fan-P&WA) (Front Frame - GE)	3 Elapsed hours 5.0 Manhours	4.2 Elapsed hours 10.0 Manhours
Turbine Rotor	9.5 Elapsed hours 31.0 Manhours	9.5 Elapsed hours 20.1 Manhours
Combustor	7.0 Elapsed hours 21.5 Manhours	9.0 Elapsed hours 22.2 Manhours

Table 8-A. Maintainability Features (Concluded)

Feature	P&W JTF17A-21B	GE4/J5P
(3) Accessibility/Packaging on-airplane		
Oil and Fuel Filters and Screens	GOOD	GOOD
Borescope ports	GOOD	GOOD+
Ignitors and Excitors	FAIR	GOOD
Bearing Replacement	GOOD	GOOD
Engine Components	GOOD	GOOD
Compressor blade replacement	FAIR	FAIR
Fuel Nozzles	FAIR	GOOD
(4) Accessibility/Packaging Off-airplane		
Bearing Replacement	GOOD	GOOD
Compressor Blade Replacement	FAIR	GOOD
Turbine Stator Vanes	GOOD	GOOD
Turbine Blade Replacement	FAIR	GOOD
Major Sections	FAIR	GOOD+
(5) On-Condition Maintenance Program		
AIDS program	FAIR	GOOD
Inspection	FAIR	GOOD
(6) Maintenance Material Costs		
(a) Rework capability	GOOD	GOOD+
(b) Dollars/flight hours	Not Stated	Not Stated

8.1.2 Evaluation

An evaluation of the maintainability features for the GE4/J5P and JTF17A-21B engines is presented in Table 8-B.

Table 8-B. Engine Maintainability Evaluation

Evaluation Item	Total Points	P&W JTF17A-21B	GE4/J5P
1. Maintenance Frequency P&WA engine has lower premature removal rate than GE engine, GE's design life goals with repair are higher by a factor of 1.2 to 1.8 than P&WA. The indication is that more repair will be required on the GE. Inspection frequencies are about the same although GE has not stated a HSI frequency. Both manufacturers are planning on on-condition repair. Inspection frequencies will have to be established to support this concept.	10	10	8

Table 8-B. Engine Maintainability Evaluation (Continued)

Evaluation Item	Total Points	P&W JTF17A-21B	GE4/J5P
2. Maintenance Manhour Goals P&WA has not provided engine maintenance manhour goals. GE has provided MMH goals. P&WA and GE have both provided elapsed time and maintenance manhour goals for individual engine components. The goals provided by GE appear to be optimistic. P&WA component goals appear realistic. P&WA was reduced 5 points for not providing total engine goals.	15	10	15
3. Accessibility/Packaging On-Airplane P&WA components have been packaged and well located for ease of removal. Pods have been provided to hold the heavy engine components for removal and replacement. GE has packaged most engine components in a module thus requiring opening of a second cover. The components in the module are difficult to remove individually with the module on the engine. The module should remove easily.	10	10	8
4. Accessibility/Packaging Off-Airplane GE has modularized the engine for replacement and repair of major engine sections and components in the shop. The P&WA engine is much the same as present P&WA engines in its modular sectioning. The GE design should reduce repair and overhaul time considerably in comparison to P&WA engine.	30	21	30
5. On-Condition Maintenance Program GE has defined on-condition determination requirements, and incorporated design features to facilitate these inspections. GE's program for the aircraft integrated data system development is well thought out. Test parameters and test points are well defined. P&WA does not have as good a program and does not appear to have done as much detailed work in either area when compared to GE.		11	15

Table 8-B. Engine Maintainability Evaluation (Concluded)

Evaluation Item	Total Points	P&W JTF17A-21B	GE GE4/J5P
6. Maintenance Material Costs Neither engine manufacturer has provided any information or data on maintenance material costs. In reviewing both engines for features which affect this cost, it is evident that GE has done considerable work to provide extra strength for reworking frames, blades, sheet metal, etc. It is noticeable in many areas that P&WA has actually lightened parts and sections over previous P&WA engines, thus reducing the reworkability of the engine.	20	3	5
TOTAL	(100 possible)	65	81

8.1.3 Conclusions
Viewing the two engine designs from a pure maintainability standpoint, the GE4/J5P engine is better than the P&WA JTF17A-21B engine. The maintenance frequencies do not appear to be drastically different. The on-airplane maintenance effort should be about equal except for internal inspections and onboard capability. The GE4/J5P has better provisions for internal inspection. The onboard monitoring of the engines could be brought to an equal capability during design and was not considered as a major discrepancy on the part of the P&WA engine. The major difference between the engines is seen in the off-airplane repair and overhaul. The GE4/J5P is superior to the P&W JTF17A-21B because of its modular construction which should result in a lower elapsed time and cost for repair and overhaul.

8.2 RELIABILITY
8.2.1 Basic Engine Design
The sections of both engines are evaluated with respect to acceptable life capability and infrequent failure potential (refer to Table 8-C). The criteria applied include the manufacturer's experience with the particular design, relative complexity, and use of design features which specifically aid reliability.

Pratt & Whitney Aircraft has the benefit of substantial experience meeting the high reliability requirements of airline engines, while the GE military engines are produced with a lower reliability factor. Overall the two offerings are rated about equal on basic engine design considerations (Table 8-C).

8.2.2 Reliability Goals and Apportionments
Pratt and Whitney Aircraft scores higher than GE for inflight shutdown and augmentation loss rates. Premature removal rates are about equal. Reliability growth also favors P&WA for early flight and mature engine goals. The time base for the early flight goals as stated is not definite. P&WA gives early flight goals while the GE goals are for the end of Phase III development. However, P&WA is given the better score because the early goals are much higher.

Evaluation of failure rate apportionments is based primarily on the credibility of the failure rates stated for the various engine components, and the relative magnitudes offered by each engine producer for the same engine sections.

The overall score under reliability apportionment on the accompanying Table 8-C favors GE because of better recognition of the difficulty of radically reducing failures as demanded by SST system reliability with such severe operating and environmental conditions. The higher overall engine failure rate offered by GE is compatible with SST system reliability.

8.2.2.1 Life
Evaluation of the design life objectives of the engine sections is based on the magnitudes of the proposed life values in relation to the severity of the respective operating conditions and environments of the engine sections. These factors are combined with the relative experience of two producers with the particular types of components. The scoring appears in Table 8-C.

8.2.3 Reliability Program
The programs of the two engine producers are basically equivalent. Both employ staff organizations which support engineering and have well defined responsibilities. The GE program activity is more extensive and makes use of sophisticated tools and procedures. The P&W program directs emphasis to areas known to have the greatest effects on engine problems in commercial service. Training is emphasized in both programs.

P&W experience in activities to attain the high reliability levels demanded by commercial airlines should aid the conduct of an effective reliability program. The GE program activity must undergo re-orientation from military to commercial environment.

Test planning, monitoring, and data feedback is systematically and effectively planned by both companies. Computerized storage and retrieval of historical reliability data appears equivalent in both programs. Both companies exhibit constructive attitudes toward the potential reliability problems in their proposed engines.

8.2.4 Reliability Evaluation Summary
The overall score in Table 9-C favors P&WA. The first component of the total score, Basic Engine Design, represents capability of the basic engine to achieve life and reliability goals with respect to manufacturer's experience, relative engine complexity and use of design features specifically to aid reliability. Pratt and Whitney Aircraft is slightly favored. The second area, Reliability Goals and Apportionments, reflects the magnitude and credibility of established failure frequency goals. Pratt and Whitney Aircraft receives 26 points to GE's 19. The third scoring component, Life, deals with the respective magnitudes of the proposed life goals in the light of severity of operating conditions in the various engine sections. The P&WA score is 12 to 10 for GE. The final area of scoring is the estimated effectiveness of two reliability programs in increasing the reliability of the engine ultimately produced. Here the companies receive equal scoring.

8.3 SAFETY

8.3.1 Engine Rotating Machinery Safety
The General Electric and Pratt & Whitney Aircraft engines proposed for the SST airplane both require containment within the engine case of failed compressor and turbine blades, stator, and guide required by FAR 33.19. The fan duct on the P&WA engine provides greater inherent energy absorption mass than is provided with the GE turbojet engine. The fan duct provides an added protection against compressor and turbine disc failures which is not part of the turbojet engine. This is not true in the fan section where the fan blades extend to the engine outer case.

The P&WA engine is shorter than the GE engine. On the B-2707 airplane, the shorter P&WA engine has an advantage in placement which results in a better placement of rotating machinery. Both engines are located with the aft end of the exhaust nozzle 50 in. aft of the horizontal stabilizer trailing edge. This places the rotating machinery on the P&WA engine aft of the main fuel tanks. To obtain the same rotating machinery location for the GE engine would result in a weight and balance penalty.

8.3.2 Engine Fire Safety
The P&WA fan engine has three inherent fire safety advantages:

(1) The engine case temperatures at the aft end of the engine cavity are 300°F less at supersonic cruise. This reduces the [illegible]ility

Table 8-C. Reliability Evaluation Summary

	GE	P&WA
BASIC ENGINE DESIGN (35 Points)		
Compressor (and Fan)	4	5
Combustor	5	4
Turbine	5	4
Augmentation	2	5
Exhaust Nozzle & Reverser	3	5
Accessories & Controls	5	4
Bearings & Seals	5	4
Subtotal	29	31
RELIABILITY GOALS & APPORTIONMENTS (30 Points)		
Overall Engine — Inflight Shutdown Rate	1	3
Augmentation Loss Rate	1	3
Premature Removal Rate	2	3
Growth — Early Flight Goals	1	3
Mature Engine Goals	1	3
Apportionment — Compressor (and Fan)	2	2
Combustor	2	1
Turbine	2	2
Augmentation	2	2
Exhaust Nozzle & Reverser	1	2
Accessories & Controls	2	1
Bearings & Seals	2	1
Subtotal	19	26
LIFE (15 Points)		
Compressor (and Fan)	1	2
Combustor	1	2
Turbine	2	1
Augmentation	1	2
Exhaust Nozzle & Reverser	2	1
Accessories & Controls	2	2
Bearings & Seals	1	2
Subtotal	10	12
RELIABILITY PROGRAM ELEMENTS (20 Points)		
Organization	2	2
Integration with Design Engineering	2	2
Experience	1	2
Coordination with Airframe Contractor	2	2
Test Planning, Monitoring, Data Feedback	2	2
Historical Data System	2	2
Prediction Analysis Methods	2	1
Failure Mode and Effect Analyses	1	2
Facilities	2	1
Attitude Toward Potential Problems	2	2
Subtotal	18	18
TOTAL SCORE	76	87

air temperature and reduces the possibility of spontaneous ignition of fuel from a ruptured line.

(2) The air flow through the cavity on the P&WA fan is less because there is no variable stator leakage into the cavity as there is on the GE turbojet. This reduces the amount of air available to support combustion.

(3) The P&WA fan is not as susceptible to engine case burn through from the primary combustor as the GE turbojet because the fan duct and fan duct air flow provide a barrier. There is, however, no equivalent barrier for the duct heater combustor which is similar in design to the primary combustor. Since the P&WA engine uses the duct heater throughout most of the flight, the advantage may be in reduced flame pressure of a burn through.

9.0 DEVELOPMENT PLAN, SCHEDULES, AND FACILITIES

The engine development test plan of each manufacturer was reviewed in terms of quality and quantity of testing, especially engine endurance testing.

General Electric's development test program is supported by a detailed breakdown of the various tests and schedules planned. Endurance testing throughout the entire program has been emphasized, with testing oriented about the SST mission profile conditions.

Pratt and Whitney Aircraft plans a program which is based on experience gained from past programs, especially the J58 engine. An extensive and detailed component test program is planned. The engine test program parallels General Electric's but offers less test hours by engine certification date. Endurance testing is planned with various types of endurance tests defined. However a summary of the endurance test hours or schedule was not given.

9.1 DEVELOPMENT TEST SCHEDULE

General Electric plans a component test program of over 300,000 hours, which includes 90,000 hours of testing on main engine components and 215,000 hours of testing on controls and accessories components, subsystems and systems.

The engine development test plan (shown in Fig. 9-1) will use 29 engines to provide 25,000 hours of testing by aircraft certification in mid-1974. Three of these engines will be run in the AEDC facility and will accumulate 750 hours of testing. An additional 12,000 to 20,000 engine hours will be accumulated during prototype flight testing.

Pratt and Whitney Aircraft plans a component test program of over 240,000 hours, which includes 90,000 hours of testing on main engine components and 150,000 hours on controls and accessories components, subsystems and systems.

The engine development test plan will use 15 engines to provide 14,500 hours of testing by engine type certification in mid-1971 and is shown in Fig. 9-2. An additional ~~23,000~~ 13000 hours will be accumulated by mid-1974, airplane certification.

A comparison of the significant milestones and test hours of each engine test plan is shown in Table 9-A.

Boeing considers the GE program to be adequate for both component and engine development testing. Endurance testing is well stressed during their program.

Boeing considers the Pratt and Whitney Aircraft program to be adequate for total component testing. The cumulative engine test hours at engine type certification is 14,500 hours and the P&WA JTF17A-21B engine will be certificated 12 months earlier.

9.2 ENDURANCE

General Electric plans extensive engine endurance testing. Over 20,000 hr of endurance testing includes: certification endurance (1,000 hr); accelerated cyclic endurance (4,000 hr); accelerated service endurance (8,000 hr); accelerated margin endurance (1,850 hr); and simulated service endurance (6,000 hr). These tests span the entire engine development program and encompass most of the test engines from early in the program through airplane certification.

During this test period, it is planned to accumulate high run times on individual engines. Under the accelerated and simulated service endurance test programs, General Electric plans to run five engines to 1,000 hr by engine certification and six engines to 2,000 hr by airplane certification.

Pratt and Whitney Aircraft plans an engine endurance program which includes the following type tests:

- Typical SST mission cycle endurance
- Low cycle fatigue testing of turbine airfoils
- Thermal fatigue cycle testing of rotating parts
- Company FTS endurance

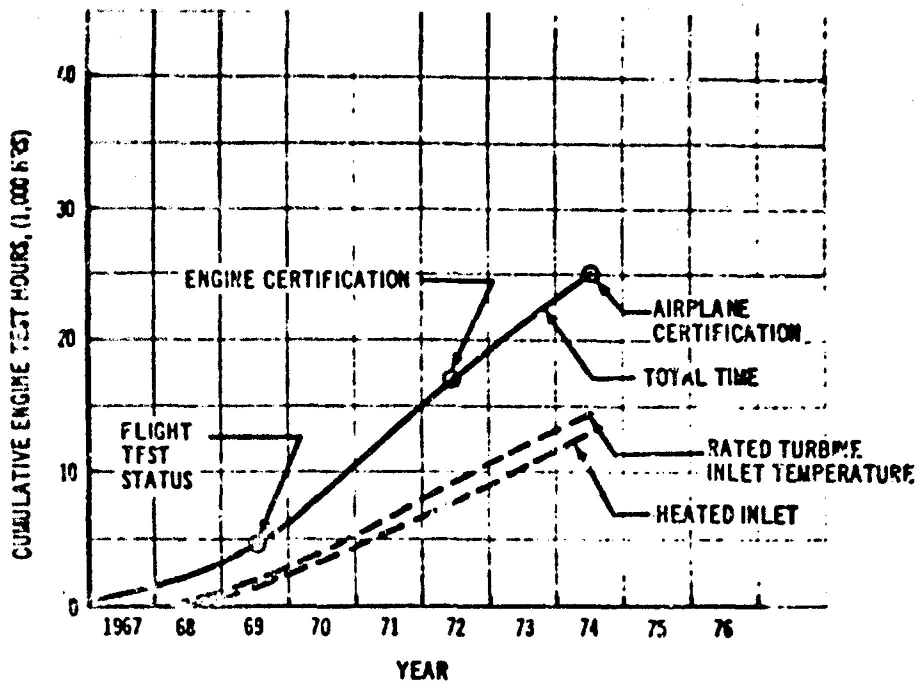

Figure 9-1. General Electric Engine Test Plan

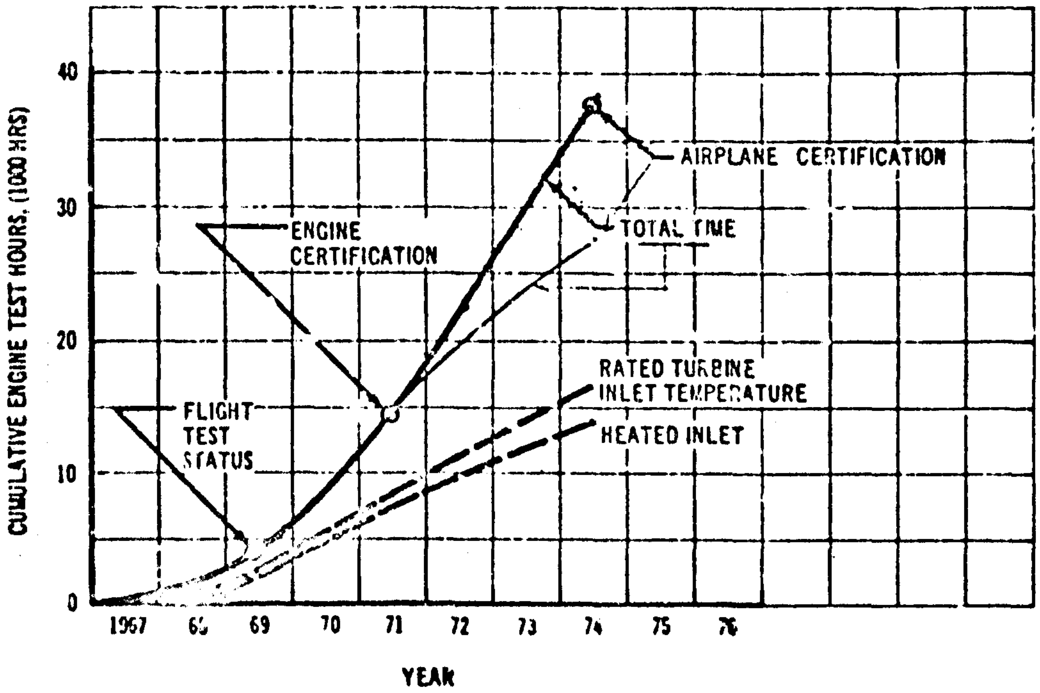

Figure 9-2. Pratt & Whitney Aircraft Engine Test Plan

Table 9-A. Engine Test Plan Comparison

	Flight Test Status (FTS)	Engine Type Certification	Airplane Certification
General Electric	(Mid-1969)	(Mid-1972)	(Mid-1974)
Total test hours	4,500	17,100	25,000
Heated inlet hours	1,600	8,000	13,090
Rated turbine inlet temperature hours	1,850	9,360	14,450
Total endurance test hours	1,950	12,500	19,650
Pratt & Whitney Aircraft	(Mid-1969)	(Mid-1971)	(Mid-1974)
Total test hours	4,000	14,500	27,500 ~~37,000~~
Heated inlet hours	2,000	7,250	13,750
Rated turbine inlet temperature hours	2,400	8,200	16,500
Total endurance test hours	1,175	2,690	---

9.3 FACILITIES

General Electric plans to use seven engine test cells located at Evendale, Ohio, to conduct their factory development program. Two of the test cells will be new ram-altitude cells to be constructed and ready for operation by January 1, 1968. An outdoor facility at Peebles, Ohio, will be used for reverser and noise suppressor development and all-weather testing. The facilities at AEDC will be used for guaranteed performance demonstrations and to conduct the inlet/engine compatibility test programs.

Pratt and Whitney Aircraft plans to use ten engine test cells to conduct their factory development program. Three of these cells presently exist; the others will be available during the period 1967-1969. Pratt and Whitney Aircraft will use the AEDC facility to conduct the inlet/engine compatibility test program. For demonstrating guaranteed performance, P&WA plans to use their own altitude facilities.

Boeing considers that GE and P&WA have provided adequate engine test facilities to accomplish their test programs.

9.4 ENGINE DELIVERY SCHEDULES

Figure 9-3 shows the engine delivery schedules for each engine. Included in the schedules are the four engines required to support the AEDC tests and the Boeing propulsion system ground rig test program. The following schedules are in consonance with the SST flight test program:

a. General Electric

General Electric has given firm delivery dates for four ground test engines with first shipping date in October 1968 and for sixteen prototype flight status engines with first shipping in July 1969. The first four prototype engines will not be flight qualified engines, but will be modified as necessary by GE after the completion of their flight qualification tests. Therefore, the first flight qualified engines will not be delivered until September 1969.

General Electric plans to have four type certified production engines available by September 1972.

b. Pratt and Whitney Aircraft

Pratt and Whitney Aircraft has given firm delivery dates for four ground test engines with first shipping date in October 1968 and for sixteen prototype flight status engines with first shipping in July 1969. All prototype engines will be flight qualified upon delivery. Four type certified production engines will be available in June 1972.

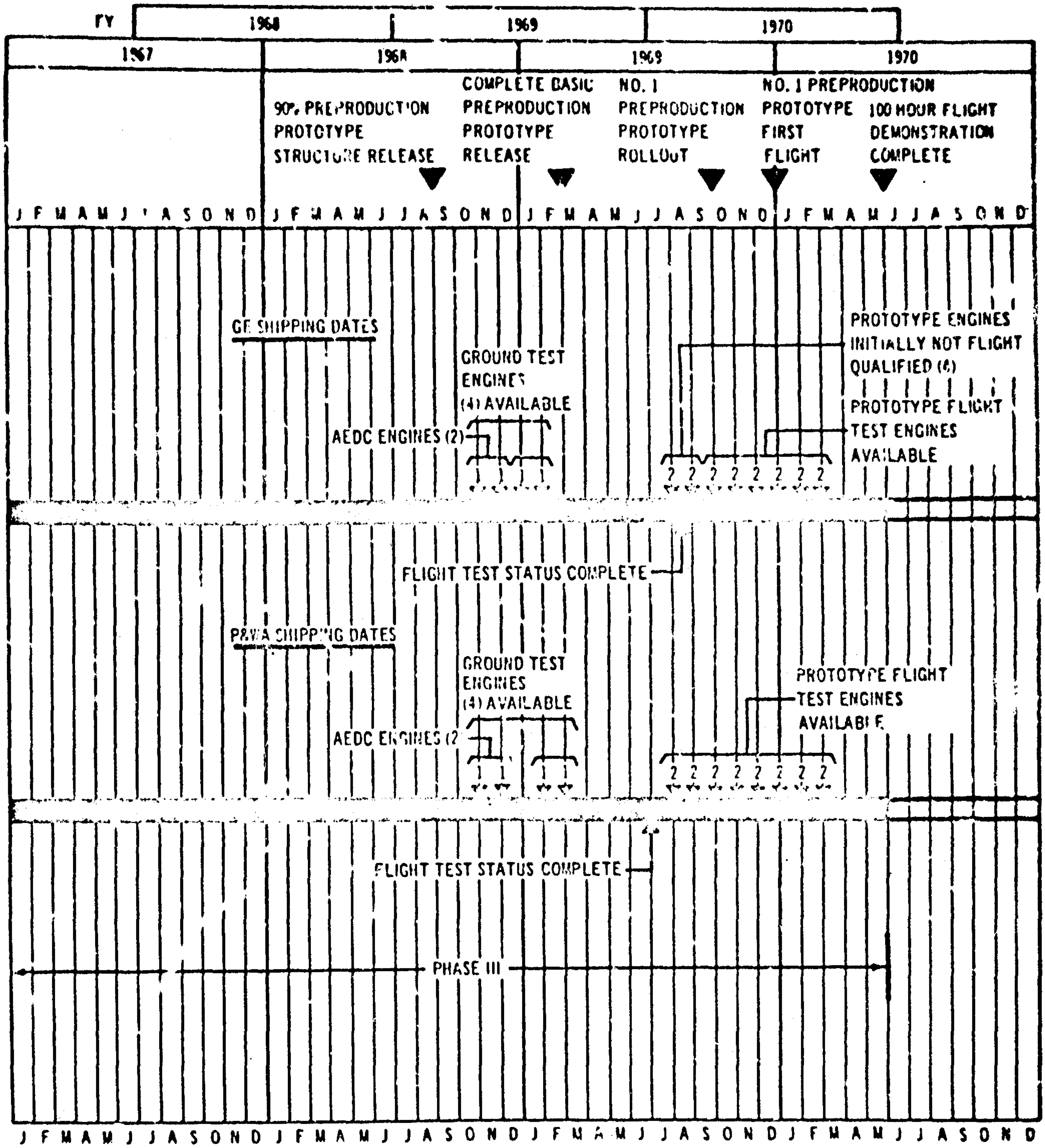

Figure 9-3. Engine Delivery Schedules

ILLUSTRATIONS

APPENDIX

ENGINE

DEVELOPMENT

RISK

APPENDIX

CONTENTS

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A1.0 INTRODUCTION

This appendix contains a technical evaluation of the component performance and mechanical design for the two SST engine offerings. Also, consideration was given to the proposed control systems, engine-inlet compatibility, and dynamic interactions of the overall propulsion system.

This evaluation is based on information from the two engine manufacturers throughout the SST Program. The most specific and recent information was obtained during a visit by Boeing representatives to the manufacturers' plants in mid-July, and from preliminary draft copies of parts of the PhaseII-C engine proposal documents.

In general, the propulsion system and components for the SST represent an advance in technology over present flight propulsion systems. Because of this, each component of the engine represents a development risk to some degree, and each component considered was placed in one of the three risk categories defined below. Briefly, component/system performance, life, complexity, and weight were considered in placing that item in a risk category. The judgement factors used were as follow:

- Design goals
- Demonstrated performance
- Past experience
- Technical capability
- Design approach

In some instances, component design and performance was, when measured against today's demonstrated technology, of such a low risk as to not warrant being placed in one of the three major risk categories; i.e., a normal development program should ensure specified performance.

The risk categories are:

a. Category 1 — The component or design has some questionable aspects at this time. Some problems are foreseen, and an above average success, in a well-run development program, will be required to accomplish the design goals. These performance goals will probably be reached.

The SST Program implications are:

- Increased development program costs
- Reduced parts life
- More complexity
- Minor program delays

b. Category 2 — A component placed in Category 2 suffers the same risk as Category 1, but to a higher degree. Moreover, additional program implications are present, especially if the goals are not reached. Quite possibly, not all goals will be reached, particularly in early commercial service.

The SST Program implications (additional to Category 1) are:

- Payload-range decrement
- Increased DOC
- Increased program delays

c. Category 3 — A Category 3 item is one which is a risk item as in (1) and (2) above, but has the potential of significantly affecting the overall program. The attainment of specified goals and performance is doubtful.

The SST Program implications (additional to Category 1 and 2) are:

- Major program delay
- Major program redirection

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A2.0 GENERAL ELECTRIC GE 4/J5P ENGINE

A2.1 COMPRESSOR

The nine-stage GE4 compressor is designed for a 12.3 to 1 pressure ratio and a mass flow of 620 lb/sec. The IGV and first stator are variable to improve performance during low inlet total temperature operation. Stators 4 through 8 are variable to improve performance during high inlet total temperature operation, and permit engine-inlet airflow matching.

The last stator is used as an aerodynamic brake. General Electric describes the compressor as a lightly loaded design.

Performance data available to substantiate the nine-stage design is from the eight-stage SST demonstration compressor designed for 475 lb/sec mass flow and a pressure ratio of 9.5 to 1. Table A-A summarizes the performance measured on the eight-stage compressor, in terms of demonstrator design goals, and measured test results.

The points shown as SLTO and CRUISE are RPM points similar to those conditions in the final engine. Both flow rate and pressure ratio exceeded design values at 100 percent design RPM. In addition to the performance tests a series of distortion tests are being run and some data are available at the present time. The data currently available for three corrected engine rotation speeds is shown in Fig. 3-14a, page 3-17, of General Electric's Volume III A proposal. Hub radial distortion up to 39.8 percent $\frac{P_{tmax} - P_{tmin}}{P_{tave}}$ was tolerated with no change in stall margin and only slight decreases (about 2 percent) in flow. The data for tip radial distortion are not available at this time, but General Electric has stated that improvement in tolerance of tip radial distortion will be designed into the nine-stage compressor.

The basic reason for changing from an eight-stage compressor to a nine-stage compressor is to increase the design flow from 475 lb/sec to 620 lb/sec. The pressure ratio overall has been raised to 12.3 to 1, but the stage loading has been decreased. Figs. 3-3a and 3-3c of Volume III C show pressure ratio and work coefficient per stage and give some indication of the decreased loading. A picture of loading is given by the Diffusion Factor (Ref. NASA RM E58D01). An examination of the relative diffusion factor levels between the eight- and nine-stage compressor has verified that loading has been reduced in every blade row except the last stator, which remained unchanged. The lower blade loading is estimated to allow stall margins to be increased and make the GE4 goals of 18 percent margin at SLTO and 30 percent margin at CRUISE become reasonably attainable. The blade aspect ratios versus hub-tip ratio have been estimated by Boeing and plotted in Fig. A-1. An increase in aspect ratio in the first rotor from about 1.3 to 2.9 is noted. This change can adversely affect compressor distortion tolerance. However, this aspect ratio is still less than that for the first rotor of the J93 compressor. An increase in mechanical tip speed has also been made from 1180 fps to 1310 fps in the nine-stage compressor.

General Electric is planning to run tests to determine the sensitivity of the nine-stage compressor to turbulence early in 1967. These tests will be similar to those conducted recently by GE and AEDC using a J93 engine.

In summation, the demonstrated test results and lack of known mechanical design problems indicate that the compressor performance closely approximates the design goals. The distortion test results reported to date on the eight-stage demonstrator compressor, and the attention being paid to both steady state and dynamic distortion, are indications that distortion problems will likely be minimized. In addition, the inherent flexibility of a variable geometry design allows significant changes in compressor performance without redesign of the compressor should problems arise. Overall, the GE compressor is not considered to be a major development risk.

Table A-A. Com.ıressor Performance Comparison

		Press. Ratio	Corr Flow	RPM Design Pc.cent	Adiabatic Eff	Stall Margin Percent	Distortion Tolerance Percent
SLTO	Design	9.51	475	100	86	17	
	Test	10	505	100	85.7	16.3	39.89*
CRUISE	Design	4.65	300	73.7	84.5	23.2	
	Test	4.65	300	73.7	84.2	19.9	24.6

* at 90 percent RPM

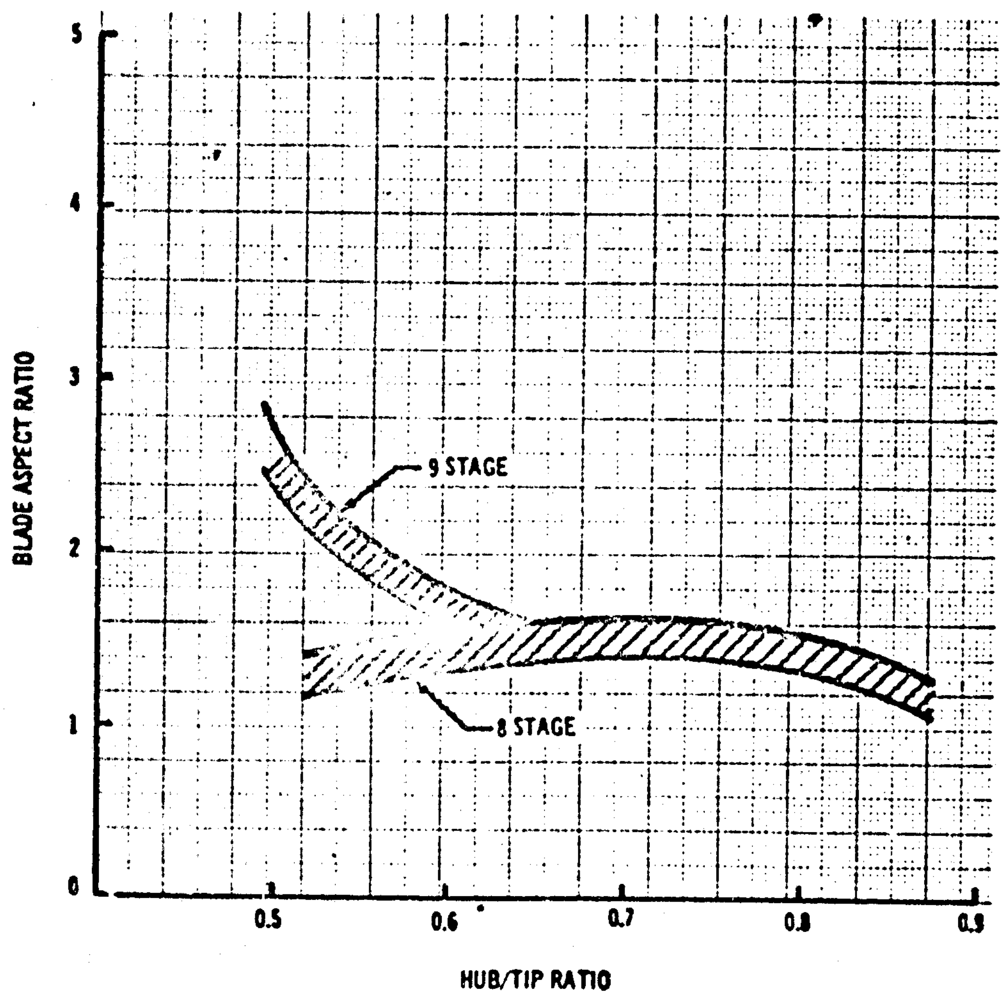

Figure A-1. Estimated Aspect Ratios Versus Hub/Tip Ratio – GE4

A1.2 MAIN BURNER

The main burner proposed by General Electric is of annular design, similar to the burner used in the J-85, T-64, T-58, GE-1, TF-39, and J-93 engines. Figure A-2 shows a comparison of the volumetric or space heat release rate for many engines. On the basis of this comparison, the space heat release rate is high, but not outside the expected growth possible for burners of this design.

Because heat release is reaction-rate limited, another parameter for evaluating gas turbine combustors is:

$$\frac{\text{(fuel flow rate) X (enthalpy increase/lb)}}{\text{(combustor volume) X (pressure)}^{1.8}}$$

with units of $\text{Btu/hr-ft}^3\text{-atm}^{1.8}$.

The corresponding value of this criterion for the GE4/J5P during Mach 2.7 cruise at 65,000 ft is 1.25 by 10^6. A limited survey shows this value to be conservative for airplane combustors and should provide margin for future growth.

The combustor thermal efficiency goal of 98.75 percent is a nominal advance over present engines. The trend curve shown in Fig. A-3 indicates that the GE combustor pressure, temperature, and reference velocity are such that high combustion efficiency values should be expected. Where heat release is reaction rate limited as in this design, combustion efficiency can be correlated with the parameter

$$\theta = \frac{P_{in}^{1.75} \times e^{(T_{in}/540)} \times A_{ref} \times D_{ref}^{0.75}}{W_A}$$

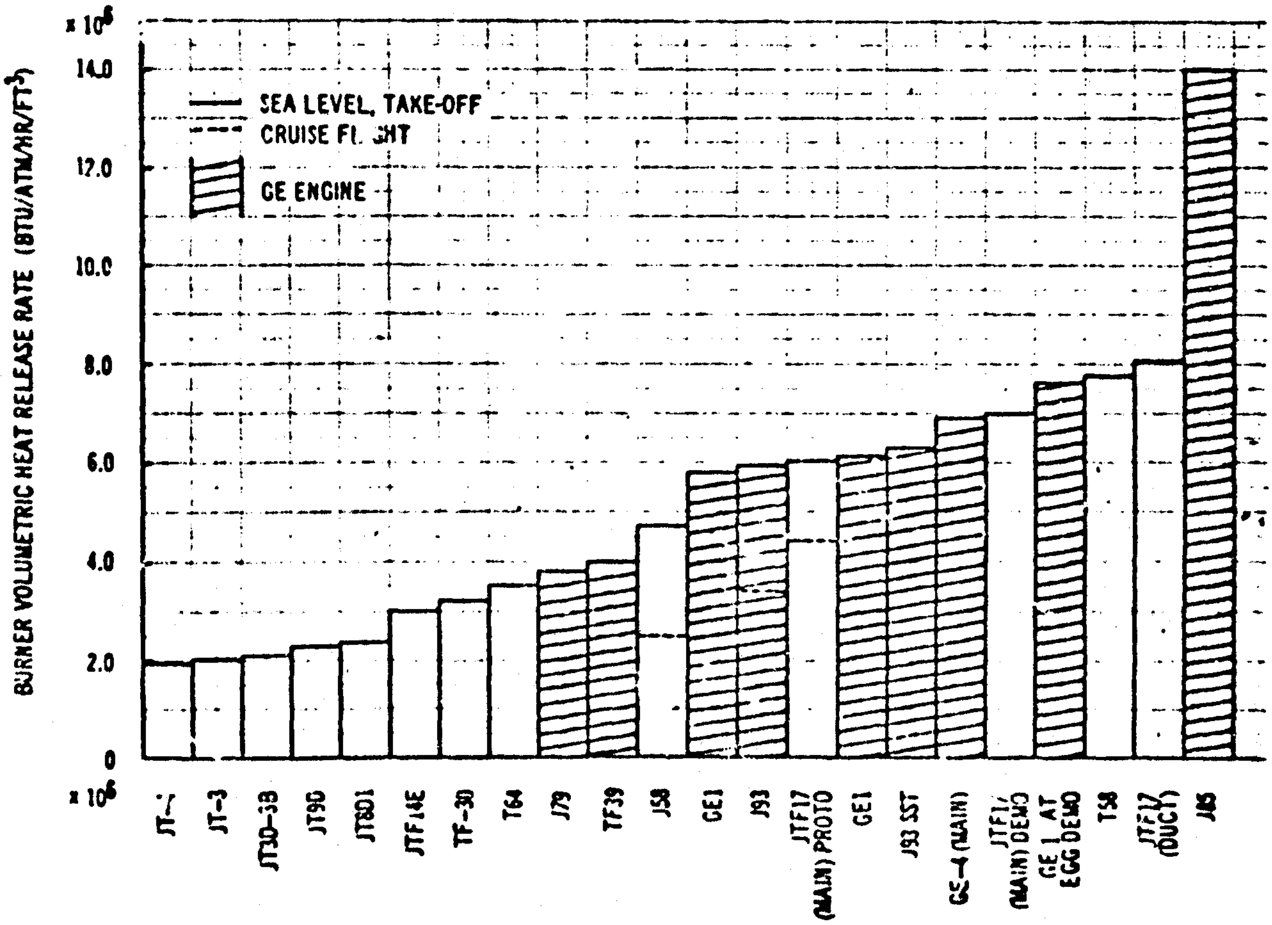

Figure A-2. Burner Volumetric Heat Release Rate – Comparison of Engine Models

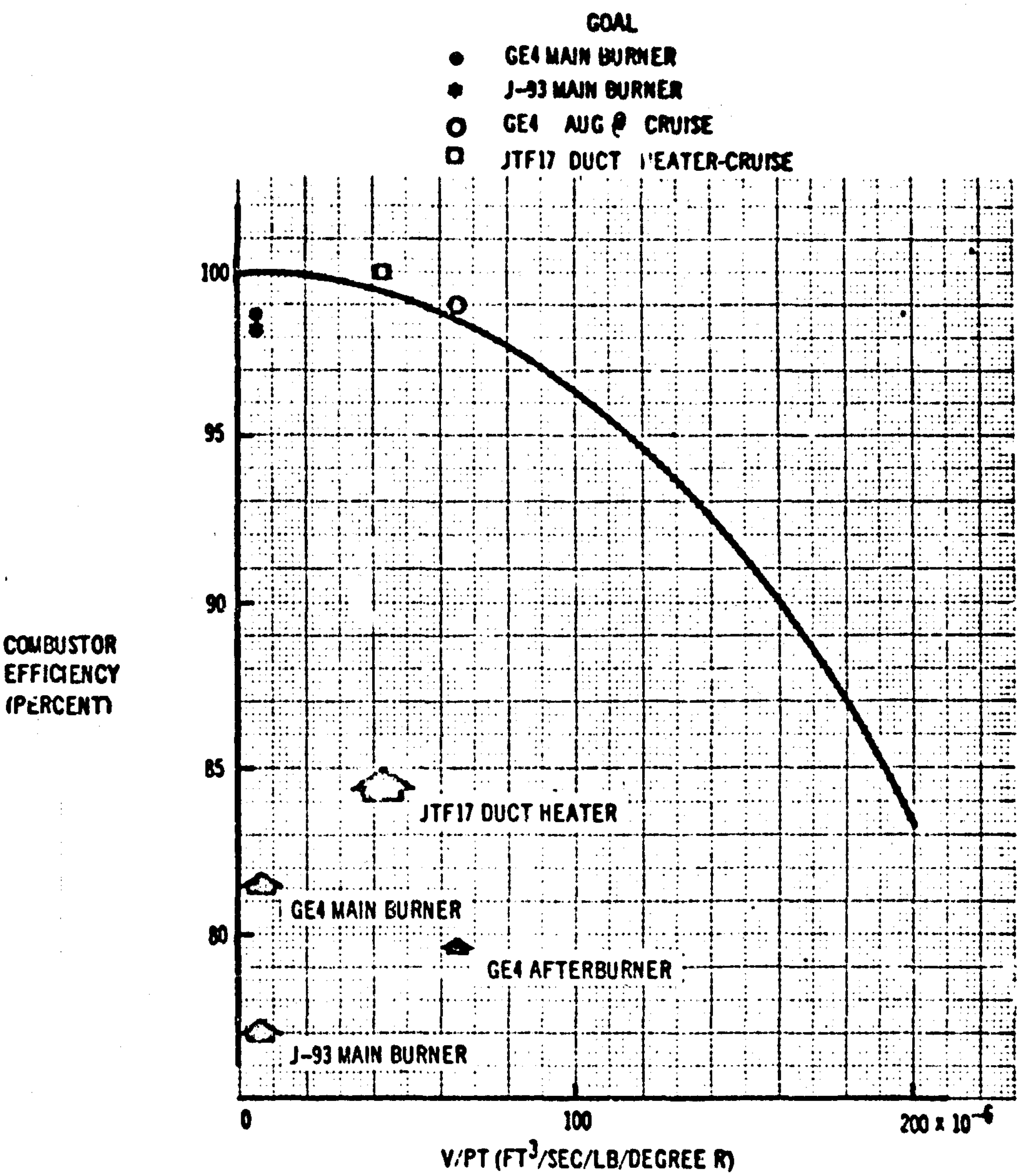

Figure A-3. Combustor Efficiency Trend

as shown in Fig. A-4 for developed annular combustors. It can be further shown that lean blowout occurs at the value of θ where the curve becomes vertical. Figure A-4 shows θ values for the GE main combustor for various flight conditions.

Burner rig testing at full-scale entry temperature, temperature rise, and reference velocity has been conducted. The one atmosphere goal efficiency value was exceeded on this test rig, but full cruise pressure testing has not been reported.

The design pattern factor is 0.20, and is considered to be a reasonable goal. The J-93 SST demonstrator engine has obtained test values of 0.16, while the TF-39 has demonstrated a value of 0.22. The temperature profile from rig testing shows a hot spot of around 100°F above

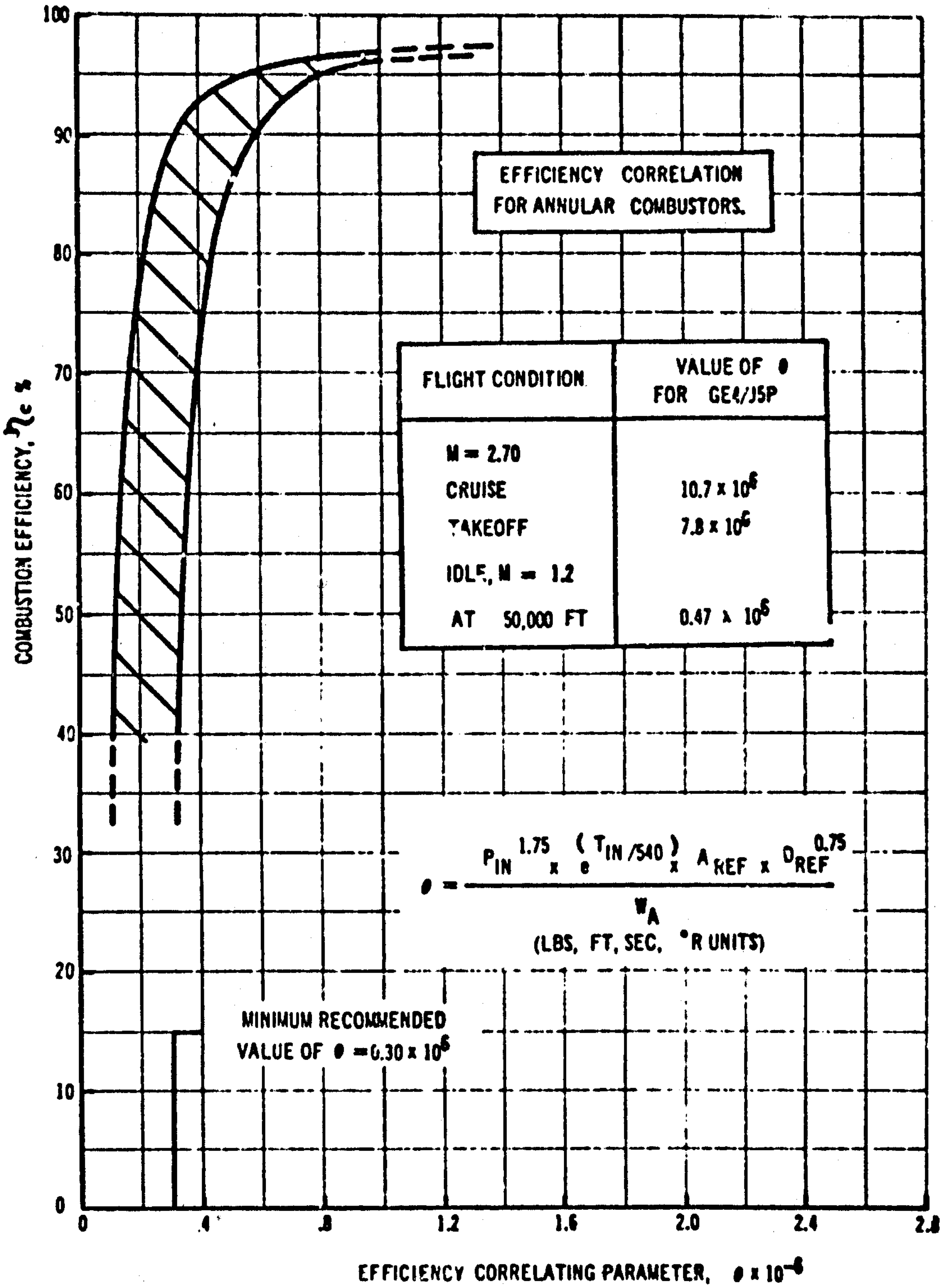

Figure A-4. GE4/J5P Combustion Efficiency Correlation

the desired value at 30 percent turbine blade height. This represents a design problem, considering the temperature profile effects on turbine blade stress, temperature, and life.

Burner rig aerodynamic distortion tests, with both radial and annular distortion, have been completed and have shown both good distortion tolerance and attenuation results.

Figure A-5 presents typical operating values of reference Mach number and pressure loss for annular aircraft-engine combustors. The design values at SLTO and Mach 2.7 cruise are shown for the GE burner. The figure indicates that the design is conservative with regard to pressure loss.

Liner metal temperatures are stated to be less than 1,500°F, with the cooling method proposed. From stress-rupture considerations, a metal temperature of 1,500°F appears to be the maximum value compatible with a long-life (10,000 hours) design. To achieve this temperature level, at the elevated turbine inlet temperatures proposed, a considerable improvement in liner cooling technique over current practice is required.

Of equal importance to the metal temperature are low metal temperature gradients. The temperature gradients shown by GE resulting from their test program are considerably lower than for the CJ805.

Mechanical design problems include welding across brazed joints, and the potential problem of rivets in the aft section of the burner separating and causing foreign object damage in the turbine.

In summary, the space heat release rates and efficiencies appear achievable but the exit temperature profiles will require further development. Overall, the GE main burner is not considered to be in one of the major development risk categories.

A2.3 TURBINE

A2.3.1 Aerodynamic Performance

The GE4 engine employs a two-stage turbine with conventional free-vortex radial aerodynamic loading distribution. The turbine aerodynamic design is summarized in Table A-B, where variables which approach severe design values have been circled for emphasis.

This turbine is a conservative aerodynamic design in the sense that all variables are within established ranges used by turbine designers.

The design offers flexibility to extract more work from the same number of stages at a small efficiency penalty, should a problem in another area in the engine establish the need for more work extraction.

The value of overall adiabatic efficiency quoted by GE is compared to other turbines on Fig. A-6 as a function of wheelspeed-loading parameter, which is a form of Parson's number.

The GE experience curve, from which the base value of efficiency for the GE turbine was obtained, is seen to be substantiated by test values for existing designs. The adjustments which GE used to get from the base value to the GE design value are also reasonable.

GE presents a detailed accounting for adverse effects of the cooling flow (including pumping work and mixing loss), balanced by the favorable effects (including boundary layer changes and increased mass flow to downstream stages), which appears to justify the small penalty for the introduction of the cooling flow.

A2.3.2 Turbine Life

The GE4 turbine employs astroloy discs with a design ultimate life, with repair, of 36,000 hours, and Rene' 69 blades with an ultimate life with repair of 12,000 hours. The discs are burst limited and therefore have a long low cycle fatigue life. The blades are creep limited.

Care must be taken in the fabrication of forged Astroloy discs to avoid undesirable low transverse ductility. GE is developing techniques in conjunction with their suppliers to enable them to avoid problems in this area.

The Rene' 69, which GE is planning to use for the turbine blades, is basically the same as IN100 (called Rene' 100 by GE) but has 5 percent more chromium which provides a material with good oxidation and sulphidation resistance. GE plans to use coated Rene' 100 blades in the first

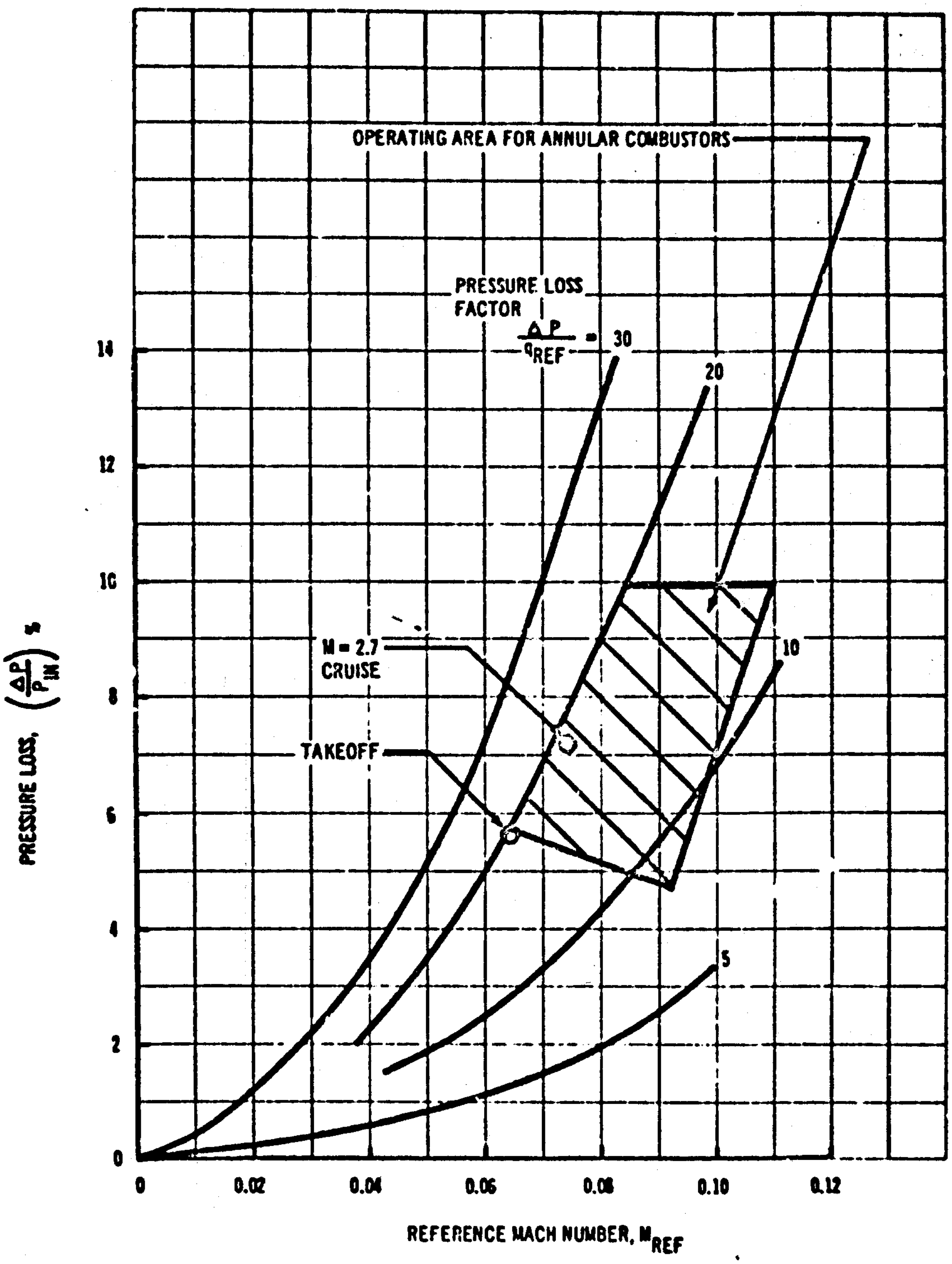

Figure A-5. GE4/J5P Burner Pressure Loss

Table A-B. Summary of Turbine Aerodynamic Design

Design Quantity	Mild Design	Severe Design	GE 4 (Free Vortex)	
			First Stage	Second Stage
At Mean Radius				
Stator exit angle β_2	30°	20°	23.14	25.91
Stator exit mach $(c/a^*)_3$	0.8	1.15	0.91	0.82
Rotor inlet relative mach $(W/a^{*\prime})_3$	0.4	0.7	0.484	0.399
Rotor exit relative mach $(W/a^{*\prime})_6$	0.8	1.15	0.74	0.73
Rotor exit axial mach	0.4	0.7	0.403	0.413
Stator aspect ratio			2.3	3.053
Rotor aspect ratio			3.008	5.204
Stator solidity (width/pitch)			1.098	1.315
Rotor solidity			1.457	1.26
Rotor exit DHUB/DTIP	0.8	0.5	0.79	0.67
At Blade Root				
Stator exit angle β_2	30°	20°	20.93	21.62
Stator exit mach $(c/a^*)_3$	0.8	1.15	1.00	0.97
Rotor inlet relative mach $(W/a^{*\prime})_3$	0.4	0.7	0.614	0.583
Rotor exit relative mach $(W/a^{*\prime})_6$	0.8	1.15	0.70	0.66
Rotor turning angle	80°	120°	109°	102°
Degree of reaction $(W_6/W_3)-1$	0.35	0	0.1	0.06
Other Design Variables				
Stage work, Δh			95.3	74.1
Stage efficiency			89.7	89.6
Mean blade speed, U_m			1130	1130
Loading parameters, $U^2/2g\ J\Delta h$			0.268	0.345
Tip Clearance			0.060"	0.080"
Tip clearance, % blade height			1.0	0.8
Tip clearance, % tip diameter			0.1	0.13

Blocked variables approach severe design

engines, while long term tests are conducted on Rene' 69 to assure that a stable composition with no sigma-phase precipitation will be achieved for production engines.

GE will use Rene' 100 if Rene' 69 turns out to be unstable because it has the same properties as Rene' 69. Use of Rene' 100, however, requires good coatings against oxidation and sulphidation. Use of coated Rene' 100 in the GE4 would necessitate more frequent blade recoating.

Considering low cycle fatigue (LCF), GE has given Boeing a detailed accounting of the procedures which they use to predict turbine-blade LCF life, taking into account temperature gradients and the entire three-dimensional stress situation. They have achieved good correlation of analysis with test results.

LCF life of the blades is predicted by GE to exceed 12,000 hours. The large number of cooling air holes in the blades does not add to FOD susceptibility because, there are no holes at the leading edge.

Internal clogging of the cooling passages is not expected to be a problem because the cooling air supply is bled at the compressor exit hub section where centrifugal forces will minimize air contamination.

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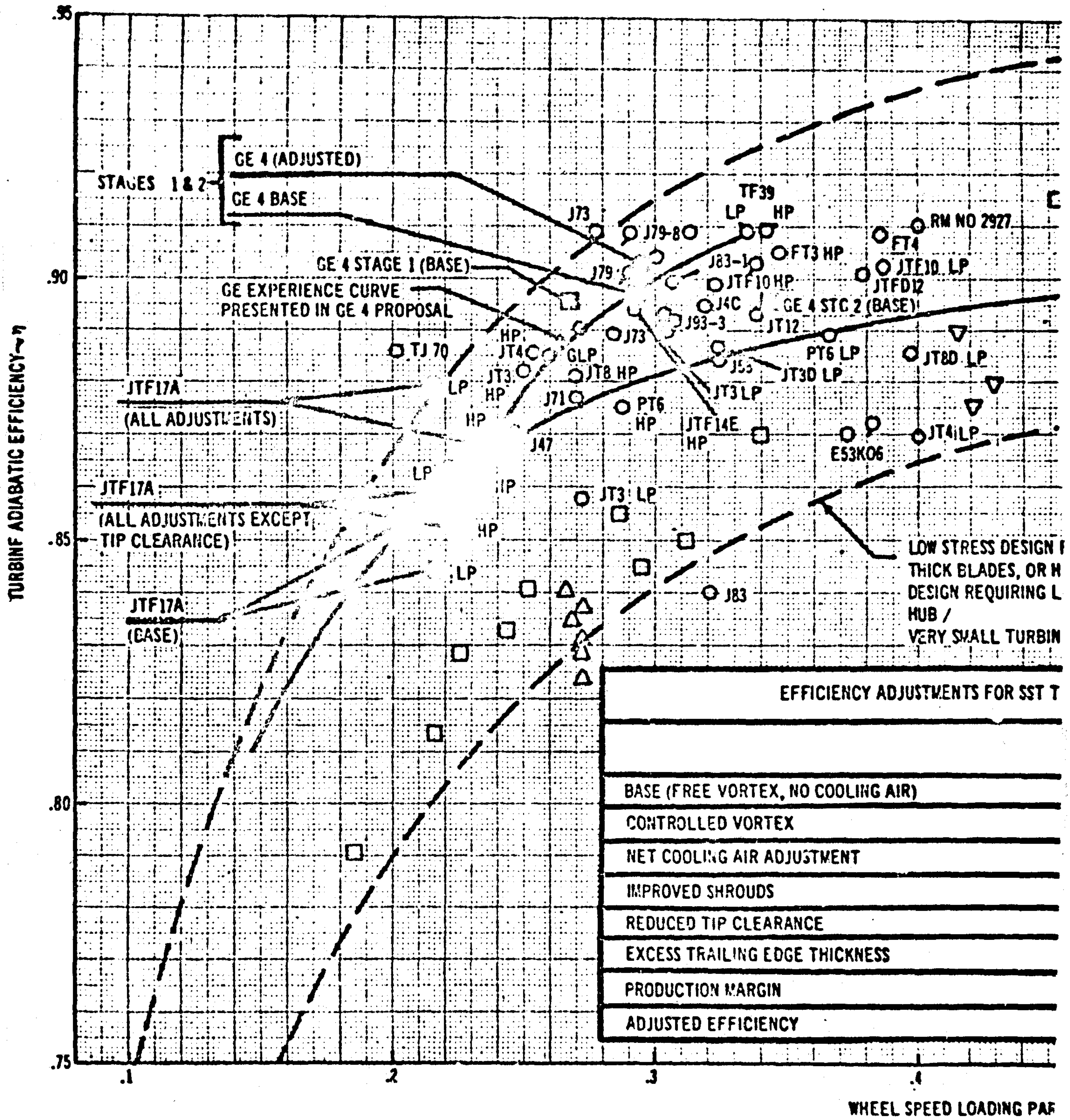

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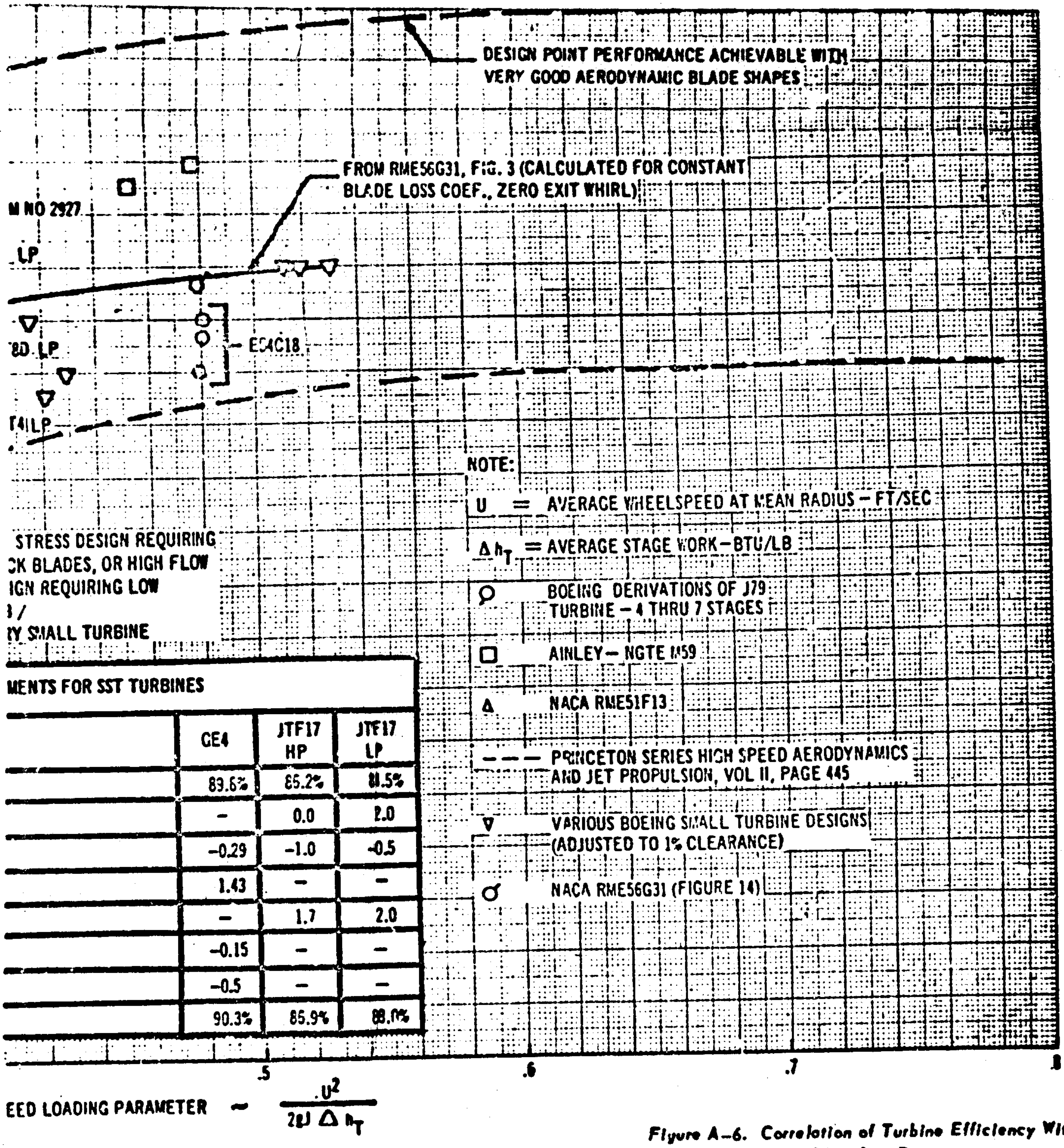

MENTS FOR SST TURBINES

	CE4	JTF17 HP	JTF17 LP
	83.6%	85.2%	81.5%
	–	0.0	2.0
	–0.29	–1.0	–0.5
	1.43	–	–
	–	1.7	2.0
	–0.15	–	–
	–0.5	–	–
	90.3%	85.9%	88.0%

Figure A-6. Correlation of Turbine Efficiency With Wheelspeed Loading Parameter

2

A simplified creep life calculation was made for the GE4 turbine to make an estimate of the risk in the turbine blade creep life quoted by GE. The mid-span section of the first rotor was chosen as a point indicative of minimum creep life. Values of temperatures and stresses were obtained from the GE4 proposal.

Several values used in the calculations are summarized as follows:

- Calculated stress = 13,300 psi
- Design factor = 0.74
- Design stress = 18,000 psi
- Average rotor gas temperature = 2,530°R
- Average metal temperature = 2,010°R
- Cooling effectiveness = 0.535
- Larson - Miller Factor = 47,800 (for Rene' 69 to 0.2 percent creep)
- Estimated creep life = 6,300 hours

The calculated life agrees reasonably well with the GE prediction, considering that a change of about 1 percent in the Larson-Miller factor changes the calculated life from 6,300 to 9,000 hours. Also, a small change in metal temperature will cause a large change in calculated life.

In summary, the GE4 turbine work output requirement is such that a two-stage turbine easily meets this requirement. The turbine is therefore a conservative aerodynamic design which should have no difficulty meeting the specified efficiency and work output. GE has other operational engines such as the J79 and J93 running at these turbine work and efficiency levels. The design offers flexibility to extract more work. For the above reasons, aerodynamic performance of the GE turbine is not considered to be a risk item. Mechanically, the first-stage turbine blades operate at a tensile stress of 13,300 psi at the critical section and have an average metal temperature of 1,550°F. Total cooling flow is 12.3 percent. There is the possibility that GE could have design problems with film cooling development in achieving the desired material properties in Rene' 69, or with coating problems if Rene' 100 is used as a substitute. The GE4 turbine blade life is therefore classified as a Category 1 risk item.

A2.4 AFTERBURNER

The GE afterburner is a direct outgrowth of the J79 and J93 augmentors. The GE4 afterburner operates in a more favorable environment, to a lower absolute temperature, and over a lower augmentation range, than its J79 and J93 predecessors. Turbine exit temperatures and therefore augmentor inflow temperature is above the auto-ignition temperature. Spark ignition is required over a small portion of the flight envelope.

The peak chemical combustion efficiency goal is 99 percent during cruise. This goal, adjusted to one-atmosphere pressure condition, has been demonstrated in a ground test rig at full-scale inflow temperature and velocity. It is not known at this time if the high level of efficiency has been demonstrated over the full cruise range of required augmentation temperatures. Referring to Fig. A-3, the combination of afterburner reference velocity, temperature, and pressure is such that the attainment of this efficiency level seems probable.

The afterburner is designed to maintain the hot flame in the center of the augmentor, thereby maintaining liner and case temperatures within design limits. No combustion takes place in the turbine discharge gas adjacent to the liner walls. In addition, compressor seal discharge leakage serves to cool the liner. The resultant liner temperature is calculated to range between 1,565°F and 1,600°F during cruise using a calculation technique verified by test. Although there is no experience under similar conditions for long life duration, the CJ805 main burner liner, under higher pressure loads, has hot spot temperatures to 2,000°F with an average temperature of 1,200°F. The time between repairs for this liner is now above 3,000 hours, with an ultimate expected life of more than 8,000 hours.

A significant life problem exists with the flameholders operating at a continuous metal temperature of close to 2,100°F. Hastalloy-X is now proposed with TD Nichrome as a more expensive backup material with 4 or 5 times

longer life. The quoted expected life for Hastelloy-X will be 2,000 hours, with repair, but long term testing to validate this goal will be required. No previous experience in commercial service exists to validate this life.

In summary, the GE augmentor is based on J79 and J93 actual flight experience. An important question is one of commercial life of augmentor parts. GE has set a design life goal of 4,000 hours without repairs. Attainment of this goal will be difficult. Because of the parts life question at this time and the consequential effects on airplane dispatchability, the augmentor life is classified as a Category 2 development risk item.

A2.5 EXHAUST NOZZLE/REVERSER

A2.5.1 GE Exhaust Nozzle

The nozzle-installed thrust minus drag coefficients proposed by GE and P&WA are plotted in Fig. A-7 as a function of flight Mach number. The values shown are from the engine companies' performance decks, with Boeing-estimated boattail drag corrections applied in the case of GE. The coefficients shown are defined on the figure and are for a typical climb and acceleration placard. Points for holding, subsonic cruise, and supersonic cruise are also shown.

The nozzle design is a new concept for General Electric.

General Electric has performed tests of nozzles similar to their offered two-stage ejector design. Model test data has been provided for their design at supersonic cruise. Data have been provided for takeoff, subsonic cruise, and Mach 1.2 climb from models quoted to be similar to the offered nozzle. These test data points are shown in Figs. A-8 to A-11. On each figure a goal point has been added, showing the installed thrust coefficient used by Boeing in its performance calculations. In each case, the GE model data meets or exceeds the specified performance. While final validation must await receipt of data from models duplicating the offered nozzle exactly, with inclusion of the effect of nozzle leakage, the GE data shown tend to substantiate their quoted performance.

Reliability and life of the nozzle and its many small parts in a hot environment represents an unknown. During afterburning, compressor discharge air is provided to keep the primary nozzle within design metal temperature limits and reduces temperature gradients. Approximately 30 hours of hot testing have been completed to date, with no problems of hinge binding.

A2.5.2 Reverser

The design of the thrust reverser is such that no major problems are anticipated in obtaining adequate reverse thrust performance with good directional control.

In summary, the lack of substantiating test data from models exactly duplicating the present two-stage ejector nozzle design, the apparent inexperience of GE with this ejector concept, and the strong effects of nozzle performance on airplane performance make the nozzle a Category 2 development risk item. The thrust reverser should perform as quoted by GE, but the life and reliability factors makes this component a Category 1 risk item.

A2.6 ENGINE WEIGHT

Boeing does not feel qualified to present a detailed analysis of the GE4 engine, but the nozzle-reverser does represent an unknown at this time. There exists no operational precedent for the nozzle-thrust reverser system as offered. Because of this, there is every reason to consider the possibility of a weight increase above the present engine company estimates. This factor, together with the consequences of overweight on airplane performance, classify the weight of the GE engine as a Category 2 development risk item.

A2.7 CONTROLS AND ENGINE DYNAMICS

A2.7.1 Control System

A functional description of the control system is presented in the following paragraphs:

A2.7.1.1 Main Fuel Control (see Fig. A-12)

a. Primary functions

Control engine speed during steady-state and transient operation of the engine.

Position the compressor variable stator vanes to achieve the required compressor air flow and stall margin.

$$\text{P\&WA INSTALLED NOZZLE COEFFICIENT} = \frac{\left(\text{EJECTOR GROSS THRUST}\right) - \left(\text{EXTERNAL DRAG}\right) - \left(\text{SECONDARY AIR RAM DRAG}\right)}{\text{IDEAL THRUST OF ENGINE AND FAN STREAM}}$$

$$\text{GE INSTALLED NOZZLE COEFFICIENT} = \frac{\left(\text{EJECTOR GROSS THRUST}\right) - \left(\text{BOATTAIL DRAG}\right) - \left(\text{SECONDARY AIR RAM DRAG}\right)}{\text{IDEAL PRIMARY THRUST}}$$

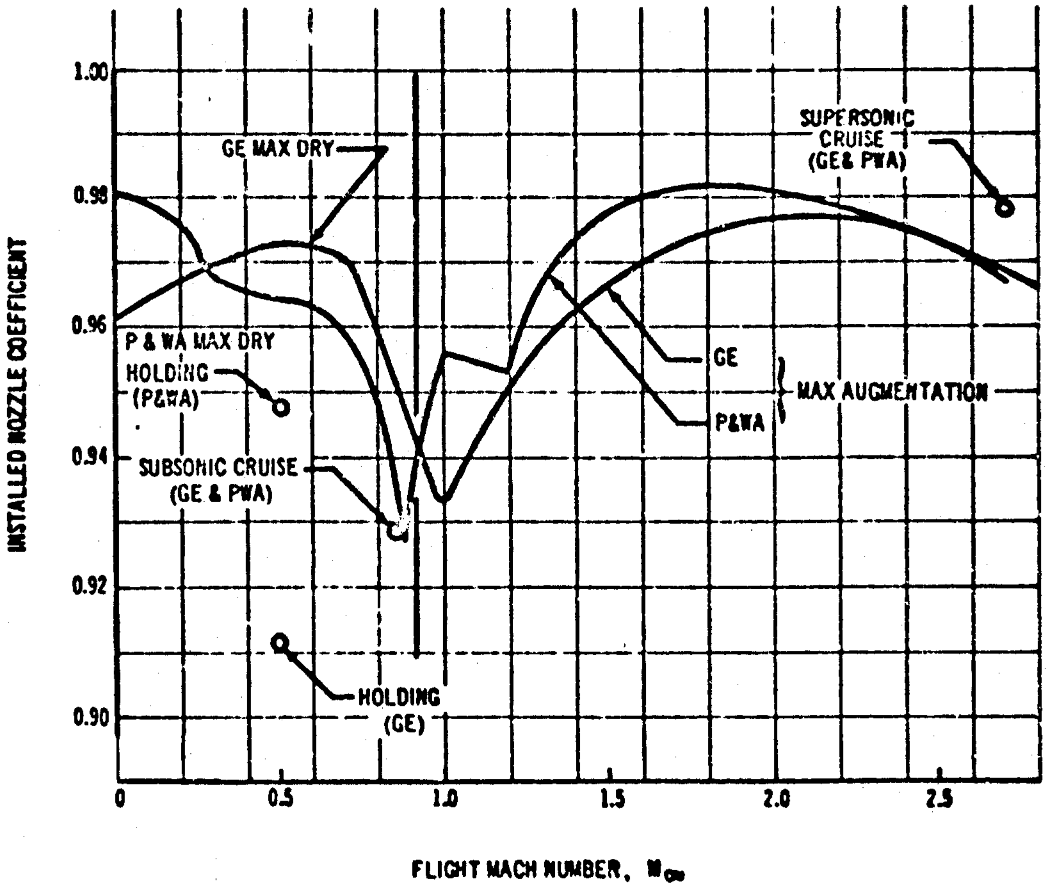

Figure A-7. Nozzle Thrust Coefficient

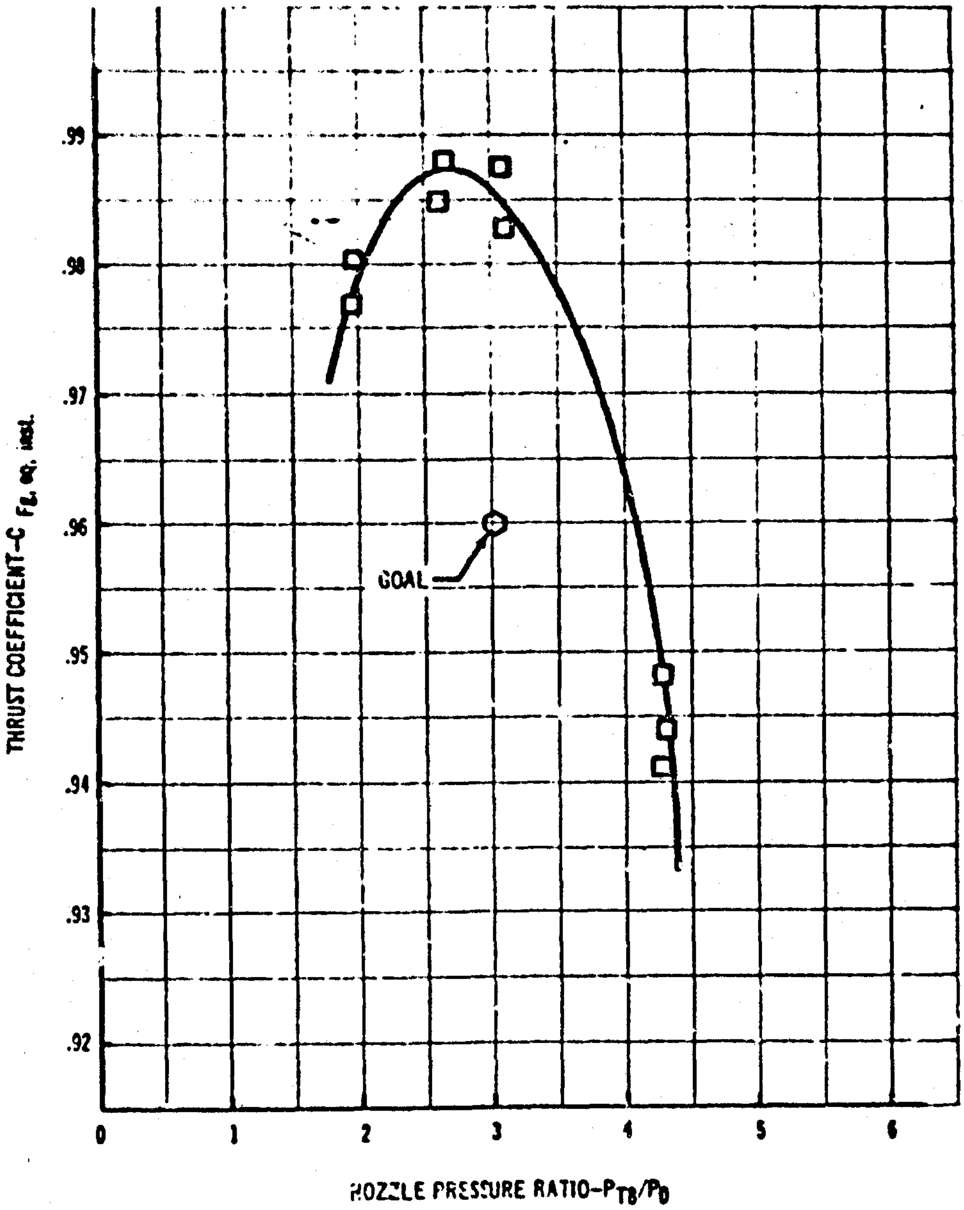

Figure A–8. Takeoff Mach=0

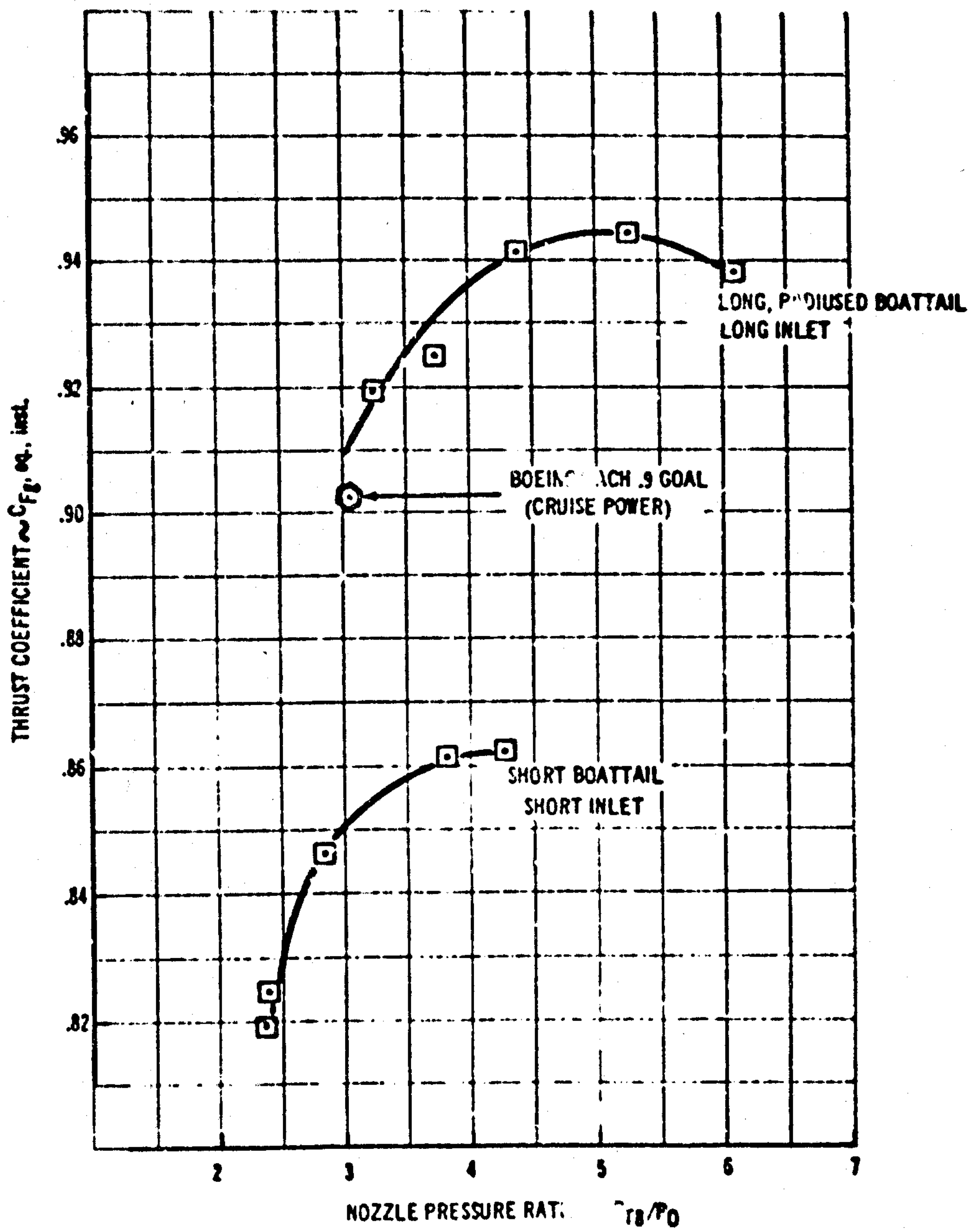

Figure A-9. Mach 0.9 Cruise

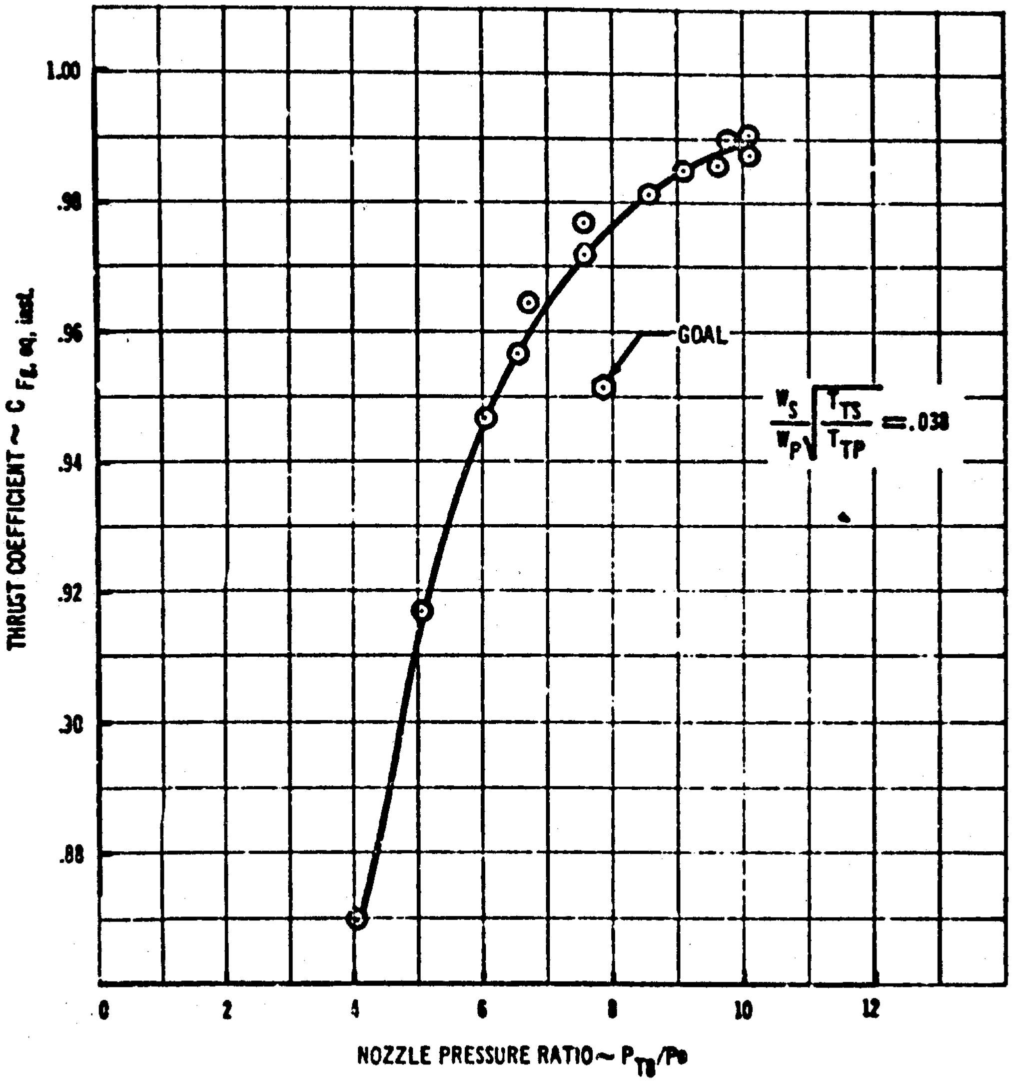

Figure A-10. Mach 1.2 Max A/B Climb

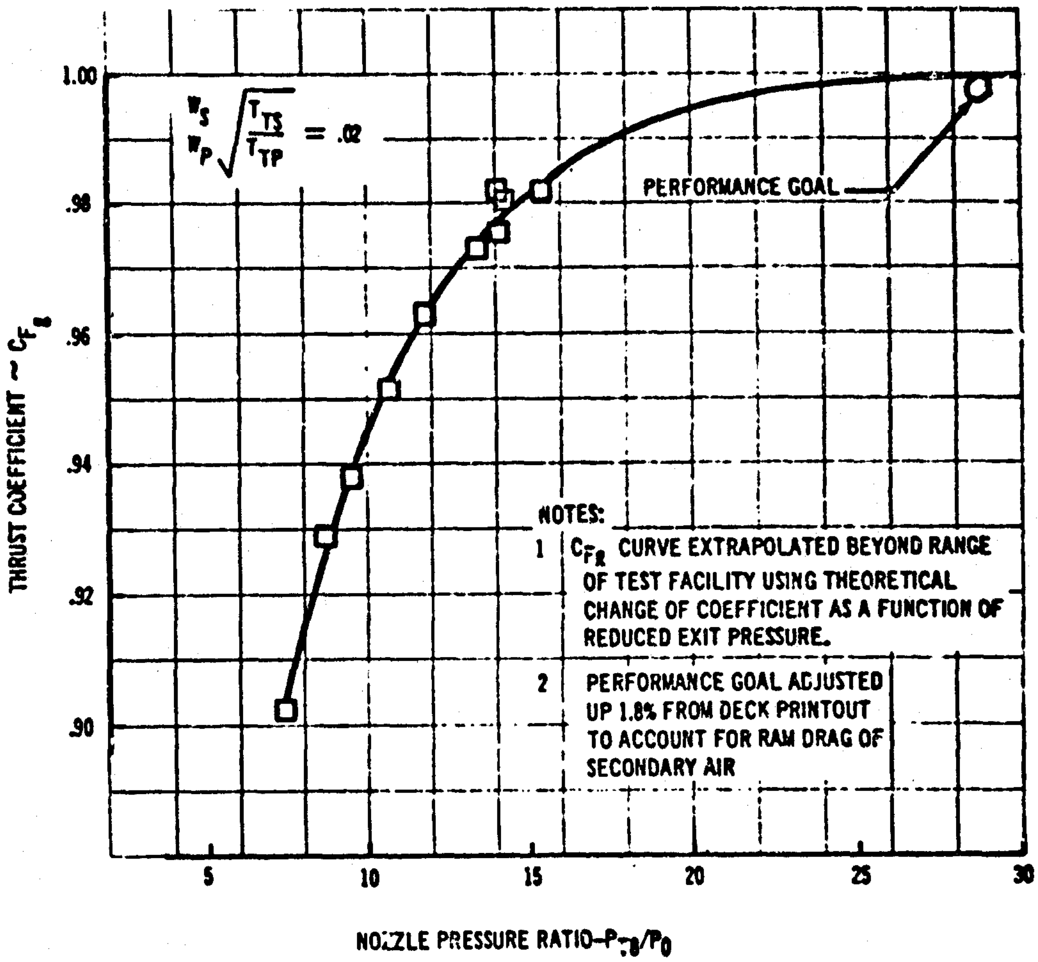

Figure A–11 Mach 2.7 Cruise

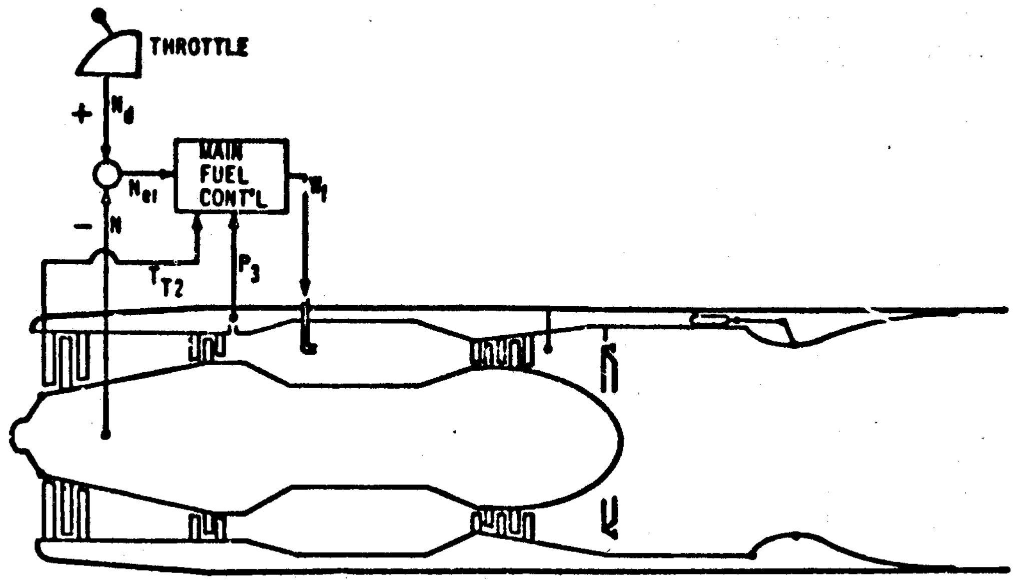

Figure A-12. Schematic Diagram of Main Fuel Control System

b. **System operation**

Steady-state

Fuel flow is varied to maintain engine rpm as required by thrust lever position. Compressor stators are positioned as a function of engine rpm (N) and compressor inlet temperature (T_{T_2}).

Transient

Fuel flow is limited by acceleration and deceleration schedules to provide rapid rpm change without encountering compressor stall, engine flame-out, or exceeding engine temperature limits. These schedules are a function of compressor discharge pressure (P_3), compressor inlet temperature (T_{T_2}) and engine rpm (N).

c. **Sensed quantities**

Engine rpm (N)

Thrust lever angle (TLA)

Compressor inlet total temperature (T_{T_2})

Compressor discharge pressure (P_3)

Stator angle (position feedback)

d. **Controlled outputs**

Gas generator fuel flow (W_F)

Compressor stator angle (2 sets $-\beta_1$ and β_2)

A.2.7.1.2 Augmentation Fuel Control (see Fig. A-13)

a. Primary functions

Schedule fuel to the augmentor during steady-state and transient operation of the engine in the augmented range.

b. System operation

Steady-state

Augmentor fuel flow is metered as a function of thrust lever angle (TLA) and compressor discharge pressure (P_3) to provide the desired temperature rise in the augmentor.

Transient

Two conditions must be met before augmentation can be initiated: thrust lever in augmentation range, and engine rpm must exceed 90 percent (to assure sufficient airflow to give successful augmentor light-off).

Following light-off, augmentor fuel flow increases until desired level is reached.

c. Sensed quantities

Thrust lever angle (TLA)

Compressor discharge pressure (P_3)

d. Controlled outputs

Augmentor fuel flow (W_{F_R})

A2.7.1.3 Nozzle Area Control (see Fig. A-14)

a. Primary function

Control primary exhaust nozzle area during steady-state and transient operation of the engine.

b. System operation

Steady-state

Primary nozzle area (A8) is varied as scheduled by thrust lever angle (TLA) from full open at idle power to nearly full closed at a power setting just below maximum dry. The variation in nozzle area is required to provide sufficient compressor stall margin at all operating conditions and allow the maximum turbine inlet temperature (T_4) to be attained at maximum dry power. The combination of increased turbine temperature (because of both rpm increase and nozzle area decrease) and increased airflow (because of increased rpm) provides thrust increase as the thrust lever moves from idle to maximum dry power.

At power settings above maximum dry (i.e., augmented power), the nozzle area must increase to allow fuel to be burned in the augmentor without increasing back pressure on the turbine. An increase in back pressure would tend to decrease engine rpm, because of decreased work output from the turbine. The main fuel control attempts to maintain 100 percent rpm during augmented operation, and gas generator fuel flow would therefore increase to restore engine rpm, thereby resulting in turbine over-temperature. To prevent this, the nozzle control senses turbine discharge temperature (T_5) and varies the nozzle area to maintain a given level of T_5. This maintains turbine inlet temperature (T_4) within acceptable limits. The reference level of T_5 is based as a function of compressor inlet total temperature (T_{T_2}).

Transient

Several features are included in the nozzle area control to improve engine transient response and augmentor light-off following throttle burst. These include:

Modifying the T_5 error signal as a function of T_5 rate of change (dT_5/dt) to compensate for thermocouple lag.

Modifying the T_5 error signal as a function of rpm rate of change (dN/dt) to improve control stability and response.

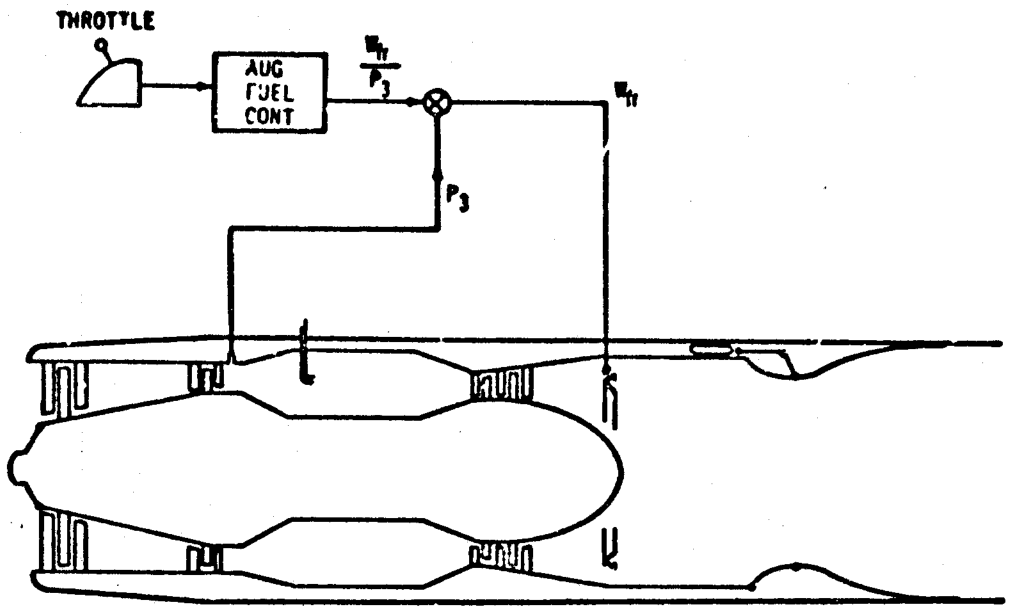

Figure A-13. Schematic Diagram of Augmentor Fuel Control System

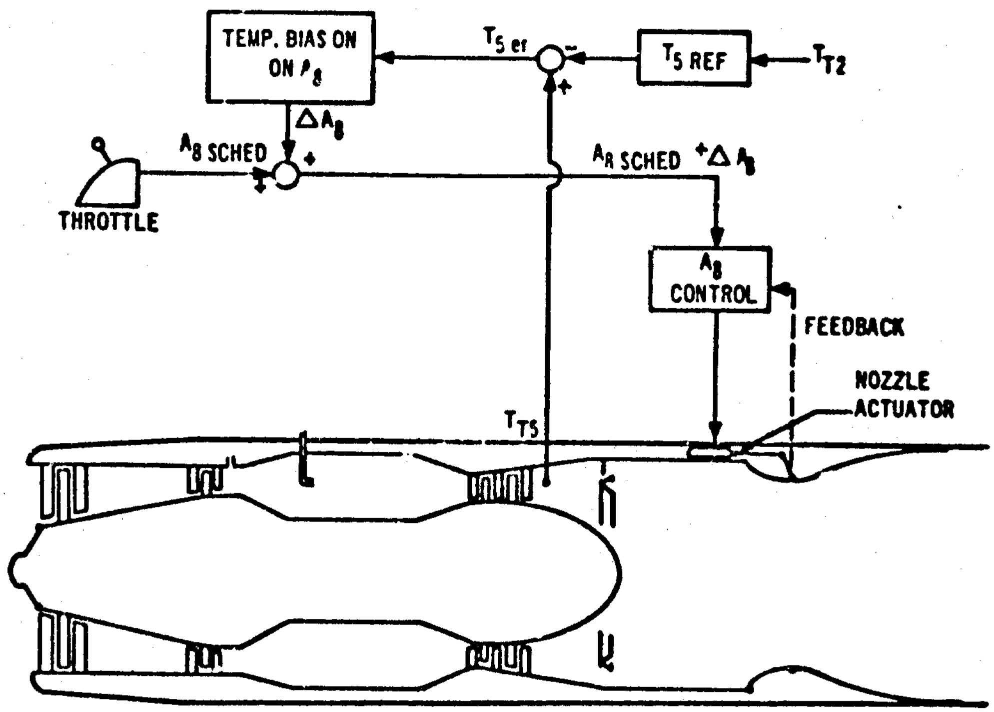

Figure A-14. Schematic Diagram of Nozzle Area Control System

Overriding the T_5 error signal as a function of rpm to hold nozzle area open until the engine is close to maximum rpm following throttle burst. Holding the nozzle open provides shorter engine acceleration times by reducing turbine back pressure during the rpm build-up.

c. Sensed quantities

Thrust lever angle (TLA)

Turbine discharge temperature (T_5)

Engine rpm (N)

Compressor inlet total temperature (T_{T2})

Nozzle area (position feedback)

d. Controlled output

Exhaust nozzle area (A_8)

A2.7.1.4 Background
The GE4 control system is very similar to the J79 (Mach 2) and the J93 (Mach 3) control systems.

A2.7.1.5 Special Features
The most significant aspect of a turbojet control is that the variable nozzle operates behind the gas generator. This gives the designer the flexibility to control gas generator parameters by means of nozzle area. Two applications in the GE4 engine are:

T_{T5} is controlled by means of nozzle area. Because there is a close relationship between T_{T5} and T_{T4} (turbine inlet temperature), there is an almost direct control of T_4 in the GE4 engine.

For operation between Idle and Maximum Dry, the nozzle is always over-area during a transient, except at 100 percent mechanical rpm. This schedule provides extra surge margin, hence faster acceleration. It also results in higher SFC below 100 percent mechanical rpm.

If GE should have control development difficulties, the design offers the opportunity to include the variable compressor stators in the control loops. In the present design, the stators are essentially used as a two-position system. However, in case of dynamic control problems, the stators could be used to vary the compressor characteristics at specific operating conditions to cope with these problems.

A2.7.1.6 Summary
GE is offering a control system which is virtually identical to previous operational control systems. The compressor variable stators offer a degree of flexibility in engine-inlet compatibility problems.

A2.7.2 Dynamics

A2.7.2.1 Background
GE in the last 5 years has worked extensively on the control system of the J93 and on a digital analog computer program (Dynasyar) for analyzing engine and control dynamics. This tool has been refined and proved on the B-70 program. Detailed dynamic effects can be studied, for example, burner blow-out can be predicted, and the varying degrees of inlet distortion during offdesign inlet operation can be studied. In summary, GE has a directly applicable background in control system simulation and has shown that this simulation gives realistic predictions.

A2.7.2.2 Simple Power Setting Changes
The two predominant characteristics of the turbojet engine which govern engine/inlet dynamics compatibility are as follows:

a. Airflow transients caused by rotor rpm changes are relatively slow because of the rotor inertia.

b. Airflow transients originating in the augmentor and nozzle are separated from the inlet by a choked turbine diaphragm.

Because these effects reduce or eliminate airflow transients, simple power setting changes will pose no problem for the inlet.

A2.7.2.3 Afterburner Light-off
Figures A-15 and A-16 show Dynasyar data for a normal light-off, and an artificially delayed (and hence extra hard) light-off. The latter case is hypothetical in that no such light-off has ever been recorded for the J93. The airflow change in Fig. A-16 falls within the Boeing inlet stability

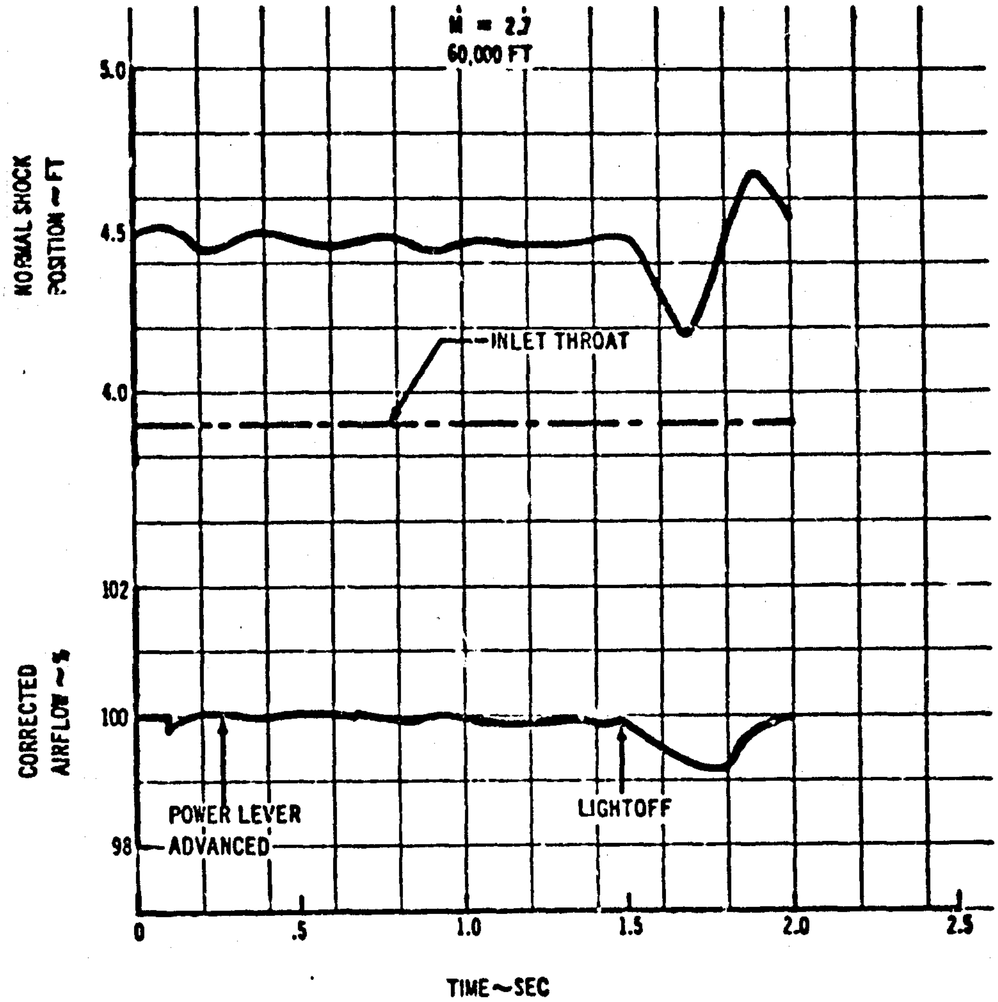

Figure A-15. GE Augmentor Normal Lightoff (Engine & Inlet)

M = 2.7
65,000 FT

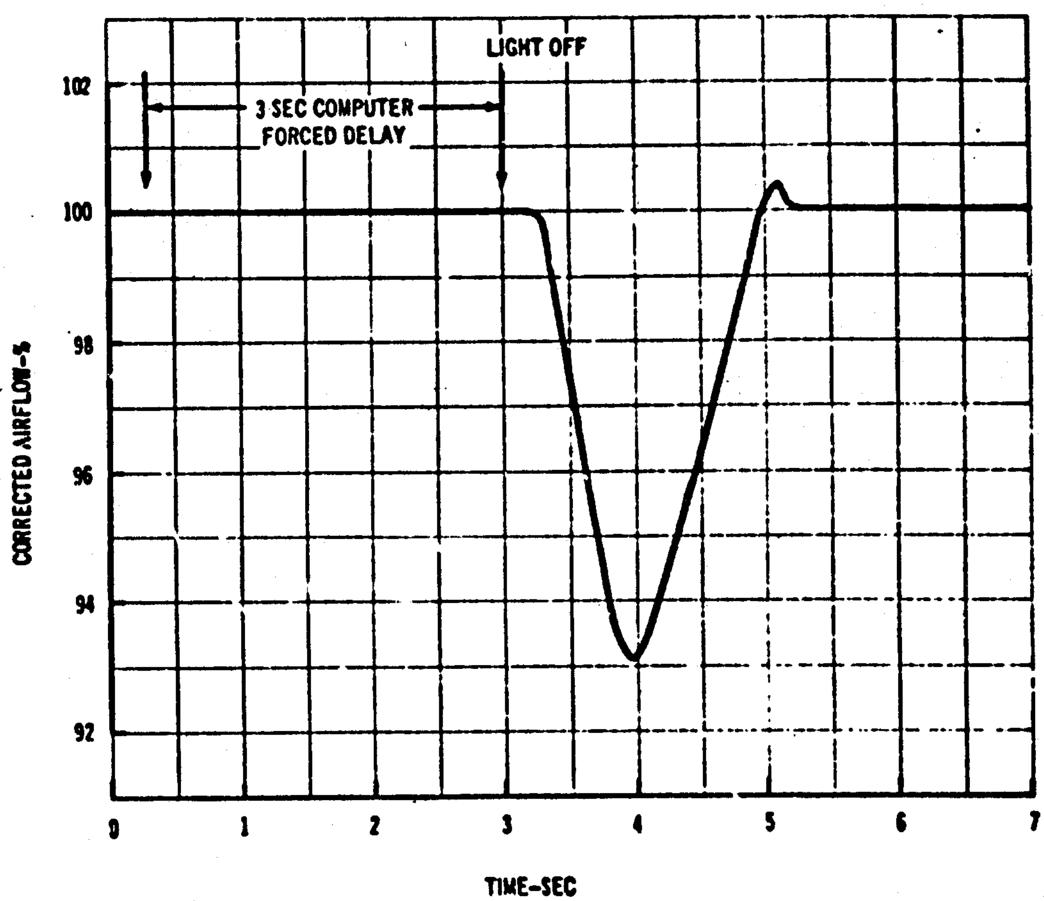

Figure A-16. GE Augmentor Hard Lightoff (Engine)

limits. These data were based on a control system without a light-off interlock. Such an interlock would (1) allow only a small fuel flow for light-off, (2) determine whether the A/B light-off has occurred by checking nozzle area change, and (3) shut off fuel flow if light-off has not occurred shortly after light-off initiation. Such an interlock is unnecessary when the augmentor operates with auto-ignition inlet gas temperatures. Figure A-17 shows that the GE4 afterburner operates during most of the normal airplane mission under auto-ignition conditions. Only in descent do auto-ignition conditions not exist.

In summary, afterburner light-off does not present inlet stability problems for the inlet, even with a delayed light-off. The afterburner operates during most of the mission under auto-ignition conditions.

A2.7.2.4 Inlet Unstart

Inlet unstart causes a sudden reduction of engine face pressure. This change increases with flight speed and is more rapid at high inlet recovery at a given flight Mach number.

J93 data for the B-70 has shown that inlet unstart at Mach 3 normally resulted in compressor surge but never in main burner flameout. The afterburner flamed out because of the initial reduction in tailpipe pressure but relit because of auto-ignition almost instantaneously. The afterburner blow-out and relight were unnoticed by the pilots.

On the B-2707 airplane, inlet unstart in cruise will cause an inlet presure transient similar in magnitude to that on the B-70. On the one hand, the B-2707 flies at lower Mach number, but on the other hand it has a much smaller volume between the shock and engine face than in the B-70. Thus the GE4 compressor may be expected to surge, but if A/B blow-out occurs, auto-ignition will restore the thrust.

In summary, the GE4 will probably experience a momentary compressor stall because of an inlet unstart. The main burner will remain lit at all times. The afterburner will probably blow out but will auto-ignite within a fraction of a second.

A2.7.2.5 Afterburner Blow-Out

Although afterburner blow-out in cruise is very unlikely, it could occur because of inlet unstart or a momentary fuel flow interruption. The latter case is shown in Fig. A-18. Relight occurs about 1 sec later and compressor airflow changes are again well within Boeing specification limits.

A2.7.2.6 Compressor Surge

The influence of the J93 experience is evident in the GE program. A continuing research compressor development program is being conducted. In addition to normal performance testing, steady-state distortion effects on airflow characteristics and surge margin are measured. Plans exist to start testing the nine-stage demonstrator compressor with turbulent and distorted inflow early in 1967. J93 experience has shown, in the last 2 years, that distortion screen testing is not sufficient. Therefore, the nonsteady distortion testing will be significant to inlet engine compatibility development.

Compressor surge caused by nozzle/augmentor transients is most improbable because of the choked turbine diaphragm between compressor and engine exhaust.

Indications are that long chord (i.e., low aspect ratio) blades are tolerant of dynamic distortion. The eight-stage demonstrator compressor has low aspect ratio (AR) blades (AR 1.3 for first stage), but the GE4/J5P nine-stage compressor will have a first stage with AR 2.88. Aft stages will have aspect ratios similar to those of the eight-stage compressor. The eight-stage compressor has been shown to have quite good steady-state distortion tolerance at the hub, but only moderate tolerance at the tip. GE is planning to distribute the steady-state distortion tolerance more evenly in the nine-stage compressor. The introduction of a first stage with AR 2.88 in the GE4 compressor makes the eight-stage demonstrator distortion data not directly applicable to the nine-stage compressor. But, eight-stage distortion characteristics indicate that GE has a good understanding of steady-state distortion. Dynamic distortion tolerance remains an open question until early 1967 tests of the nine-stage compressor.

The possible use of variable stators for solution of dynamic problems has already been pointed out. It is a high response method which has been used on some fighter airplanes to prevent compressor stall during transients (e.g. hot gas ingestion from guns). Another possibility to resolve dynamic problems is compressor redesign

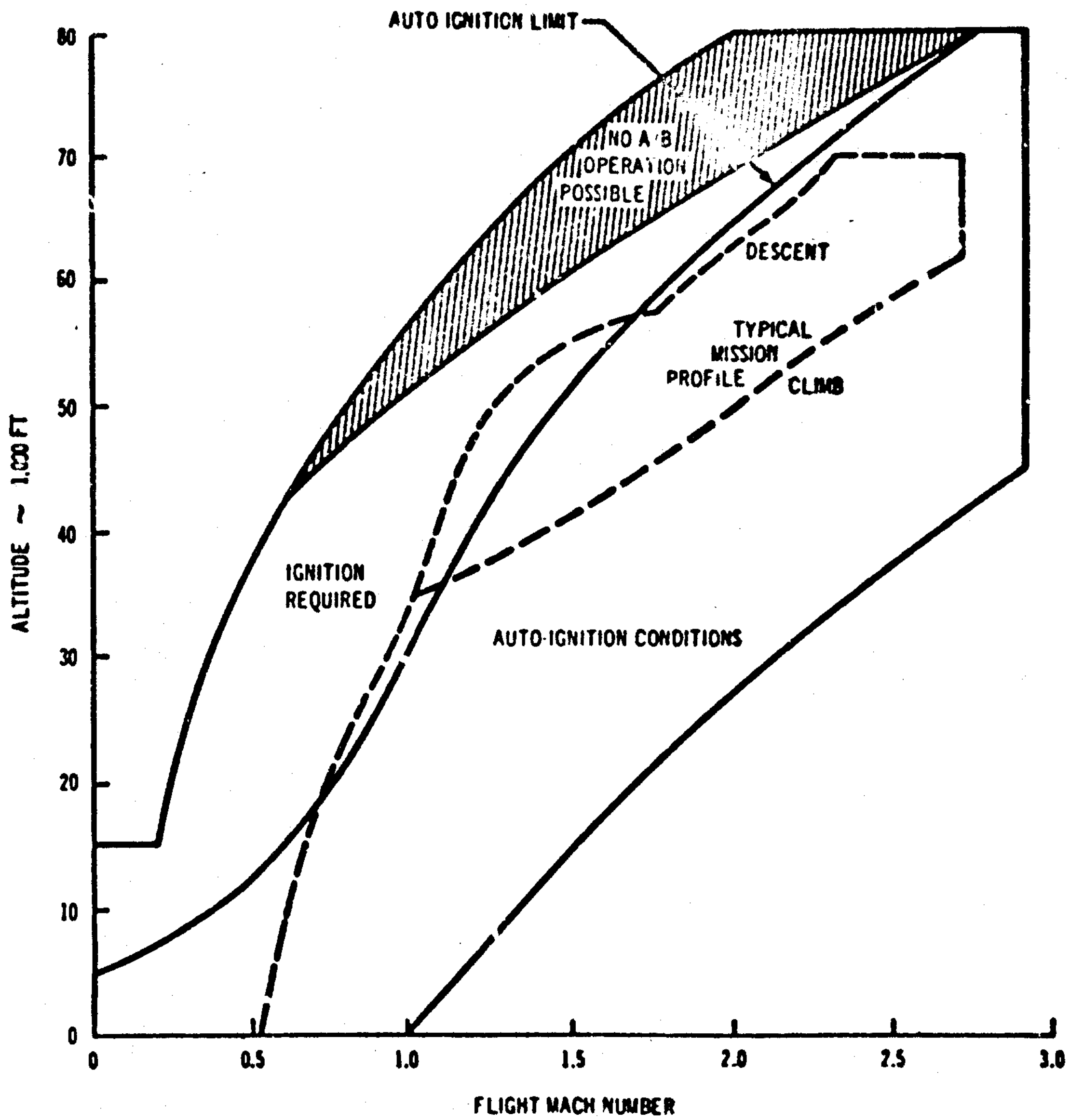

Figure A-17. GE Augmentor Ignition

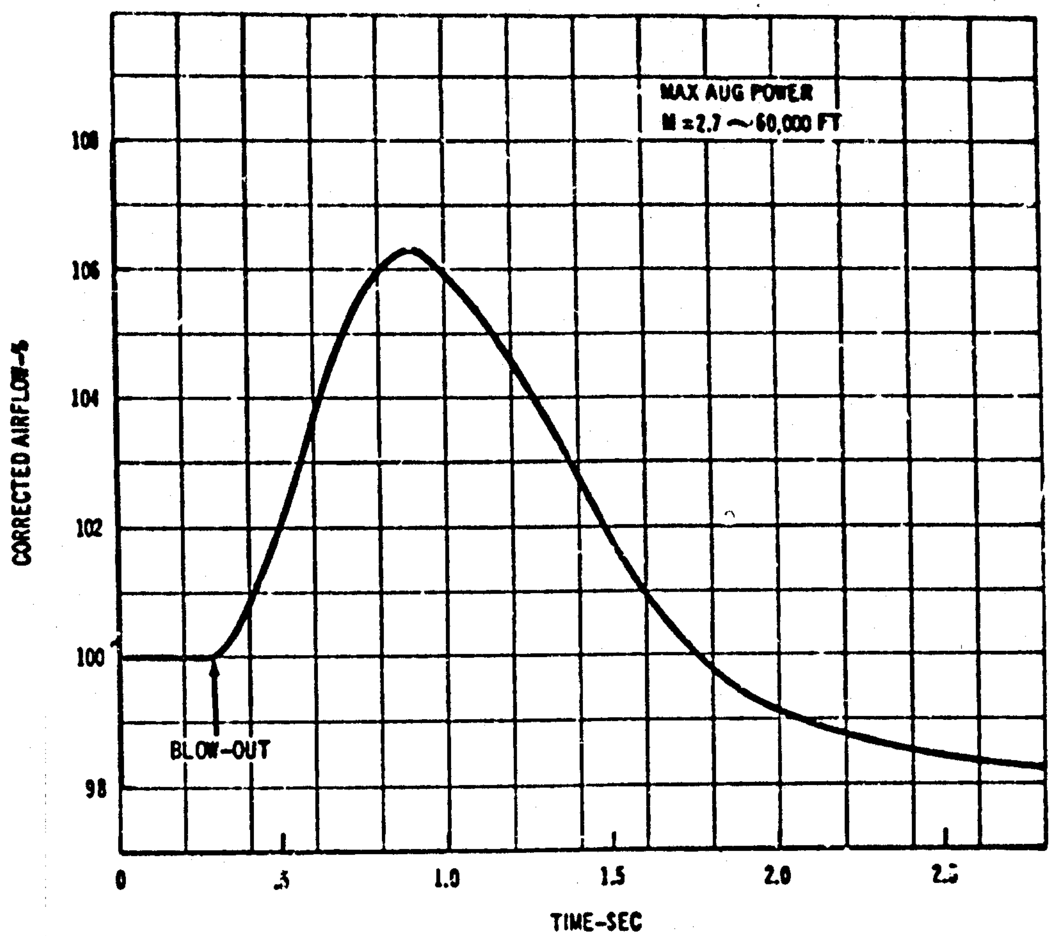

Figure A-18. GE Augmentor Blowout

to obtain lower aspect ratio blades or more stages. This would be a major undertaking and result in a weight and length penalty. However, the three-bearing design of the engine at least offers this possibility without any further design changes.

In summary, General Electric has initiated a well-defined program on compressor surge margin and distortion tolerance. Distortion test data for the eight-stage demonstrator compressor looks good for this phase of the program. However, the nine-stage compressor differs significantly from the eight-stage demonstrator. The nine-stage compressor has a first stage with an aspect ratio of 2.9 and the eight-stage compressor has an aspect ratio of 1.3. The distortion tolerance of the nine-stage compressor remains to be demonstrated. In case of persistent distortion or dynamic control problems, GE can incorporate the variable stators in the high response part of the control system.

A2.7.3 Controls and Dynamics Summary
Overall, the controls and dynamics of the GE4 engine constitute a Category 1 development risk item.

A3.0 PRATT AND WHITNEY AIRCRAFT JTF17A-21B ENGINE

A3.1 FAN/COMPRESSOR

Compression of the air for the JTF17 engine is accomplished by a two-stage fan with a 2.84 average pressure ratio and a six-stage high-pressure compressor with a 4.84 pressure ratio. The gas generator flow of 287 lb/sec at sea level has an overall pressure ratio of 12.97 to 1; the duct flow of 390 lb/sec has a pressure ratio of 2.96 to 1. The fan and compressor are treated separately in this section.

A 3.1.1 Fan

The fan flow is divided into engine side and duct side flow with fan pressure ratios of 2.68 and 2.96 respectively. The mean inlet axial Mach number to the fan is 0.592, and the tip velocity of the first rotor is 1,694 ft/sec giving a tip relative Mach number of 1.68. The fan rotors are provided with two vibration dampers each. The duct side exit guide vanes are slotted airfoil sections. The two stages currently proposed in the JTF17 fan consist of a scaled JTF14 fan as the first stage, and a second stage similar to fan build No. five tested in the P&WA 0.62-scale test rig. Based on the information available from P&WA, an attempt has been made to understand the details and proper interpretation of the fan test progress. There has apparently been a total of nine different builds with multiple designs of both first and second stage fan rotors. The following lists the major tests and findings as Boeing understands them.

Build No. 1 As designed; airflow 3-1/2 percent below design; bypass ratio 1.37 compared to 1.3 desired; pressure ratio 2.51 compared to 2.7 desired; no efficiency data.

Build No. 2 Streamlined part-span shrouds; airflow 1 percent below design; bypass ratio 1.15, 1.3 desired; pressure ratio 2.68, 2.7 desired; peak efficiency engine side 84 percent; duct side 76.5 percent.

Build No. 3 Overcambered first fan blade; airflow 1-1/2 percent above design; bypass ratio 1.15, 1.3 desired; pressure ratio 2.76, 2.7 desired; efficiencies same as No. 2.

Build No. 4 Identical to No. 2 except the outer part-span shrouds were removed from second stage blades; short test to determine performance and vibration effects of second stage blade change; high blade stresses attributed to blade flutter at 59 percent design speed; no data obtained.

Build No. 5 Identical to No. 2 except airflow splitter was drooped; same as first experimental engine; airflow 1/2 percent above design; bypass ratio 1.37, 1.3 desired; pressure ratio 2.72, 2.7 desired; peak efficiency engine side 82.5 percent, duct side 78.5 percent; significant improvement in surge margin at 100 percent speed over No. 2; separation noted on splitter; complete map obtained.

Build No. 6 Filler ring added to smooth inboard side of splitter; complete speed lines run; overall performance slightly degraded from No. 5; Stator 1 closed 4 degrees and speed lines rerun; no improvement in part speed surge; engine side flow down 1.9 percent; total airflow down 1 percent.

Build No. 7 Redesigned first and second stage blades based on No. 3 data; overcambered leading edges; relocated part-span shrouds at 75 percent and 40 percent span as against 88 percent and 50 percent span; peak efficiency exceeded cruise goal by one count.

Build No. 8 No data available.

Build No. 9 Redesigned second rotor blades, new flow splitter, will use first stage blades from No. 7.

Data from Build 5 (tested in Feb. 1966) are cited in the draft copy of the P&WA proposal as Phase II-C demonstrated values. The performance data for Build 5 are summarized in Tables A-C and A-D. Performance goals are shown for comparison. Build 5 test goals have not been clearly stated; therefore, production engine goals have been used as indicated.

Build 5 performed well in terms of flow and pressure ratio at SLTO-simulated conditions, but an improvement in engine side efficiency is required. At simulated cruise conditions, the efficiency was acceptable and the pressure ratio was reasonably close, but higher RPM was required. Stall margin on both sides of the fan at SLTO is low. The combination of a JTF14 first stage and a Build 5 second stage can certainly not be taken as final, and additional developmental changes are anticipated before desired performance goals are reached. P&WA has not provided distortion test data; however, they have stated that: "As development proceeds with the selection of an airframe contractor and further definition of the inlet design, more extensive information on the distortion will become available making it possible to start fan component testing with simulated distortions." Adequate fan toleration of distortion at the hub because of the flow inducing nature of a two-spool engine may be realized. Likewise, attenuation of distortion through the duct side of the fan may result from suitable airfoil sections, but there is currently no known way of analytically predicting these facts, and no substantiating test data is available.

In summary, the fan is in an early development state and performance goals have not been demonstrated. The lack of distortion testing together with the fluid state of the fan design make the achievement of stated fan performance goals a Category 1 development risk.

Table A-C. Fan Performance Comparison — Duct Side

		Press. Ratio	Specific Flow	RPM Percent Design	Adiabatic Eff (%)	Stall Margin (%)	Distortion Tolerance
SLTO	Goal	2.7		100	78.8*	11.0*	
	Test	2.7	41	99	78	5	
CRUISE	Goal	1.56*		63.6	89.8*	28.5*	
	Test	1.50	30.3	69	78	24	

*Production Engine Goal

Table A-D. Fan Performance Comparison — Engine Side

		Press. Ratio	Specific Flow	RPM Percent Design	Adiabatic Eff (%)	Stall Margin (%)	Distortion Tolerance
SLTO	Goal	2.7		100	88.8*	9.2*	
	Test	2.5	41	99	79	7	
CRUISE	Goal	1.56*		63.6	89.8*		
	Test	150	30.3	69	85.5	28	

*Production Engine Goal

A3.1.2 High-Pressure Compressor
Demonstrated performance for the high-pressure compressor comes from Build 5 of the 650 lb/sec compressor and is summarized in Table A-E. In the absence of specific Build 5 performance, the production engine goals have been used as indicated. Low efficiency and lack of stall margins at SLTO are definite problem areas. The stage by stage performance detailed on pages BIIA-44 and 45 of the P&WA Phase II-C draft proposal received by Boeing on 9 August 1966 represents a complete change from the stage by stage data received on a visit to Pratt and Whitney on 22 July 1966. However, no changes in overall high pressure compressor performance were noted.

In Fig. A-19, rotor blade aspect ratios are plotted versus hub to tip ratios. Two sets of data provided by P&WA earlier in the year are shown, but no data of this type is present in the current draft proposal. The plotted data were current in March and July of 1966. In the same time period, blade failures on the first rotor of HP compressor were experienced; first, in Build 1 on the compressor rig and later in the second demonstrator engine. Because of blade failures and flutter problems encountered on the C5-A demonstrator engine, the commercial version (the JT9D) has an additional stage and lower aspect ratios. For these reasons, the structural integrity of the JTF17 short chord design is now an unknown. A second unknown is the probability of achieving the necessary flow range with the short chord design approach currently being used. Several publications (for example, NACA RM E47I03, Feb., 1958) show that the flow range of a rotor definitely decreases with the decreasing chord length.

In summary, the HP compressor design is in an early development state; performance goals have not been demonstrated, and distortion test results are not available. These factors indicate that the high pressure compressor is a Category 1 development risk. Considering that the high pressure compressor must be developed to accept fan hub flow, the overall compression section of the JTF17 is viewed as a Category 2 development risk.

A3.2 MAIN BURNER
The P&WA ram induction burner is a new concept with major features being short length and high efficiency. The reduced length is made possible by a low level of diffusion upstream of the burner. Air is injected into the burner liner by a velocity head conversion rather than a static pressure difference.

Figure A-2 shows the volumetric heat release for this burner to be reasonable, with a demonstrated value higher than the design goal for the prototype engine.

The heat release parameter discussed in Par. A2.2

$$\frac{(\text{fuel flow rate}) \times (\text{enthalpy increase/lb})}{(\text{combustor volume}) \times (\text{pressure})^{1.8}}$$

for the P&WA JTF17A at Mach 2.7 cruise is 1.13×10^6. This value is considered to be conservative and should provide margin for future growth.

Table A-E. JTF17-High Pressure Compressor Performance Comparison

		Press. Ratio	Corr Flow	RPM Percent Design	Adiabatic Eff (%)	Stall Margin (%)	Distortion Margin
SLTO	Goal	4.77	130.3	100	85.9*	17*	
	Test	4.75	131.5	100	81.8	0	
CRUISE	Goal	2.92*	98.4	81	86.8*	30*	
	Test	2.75	98.4	80	85.5	27.5	

*Production engine design goal.

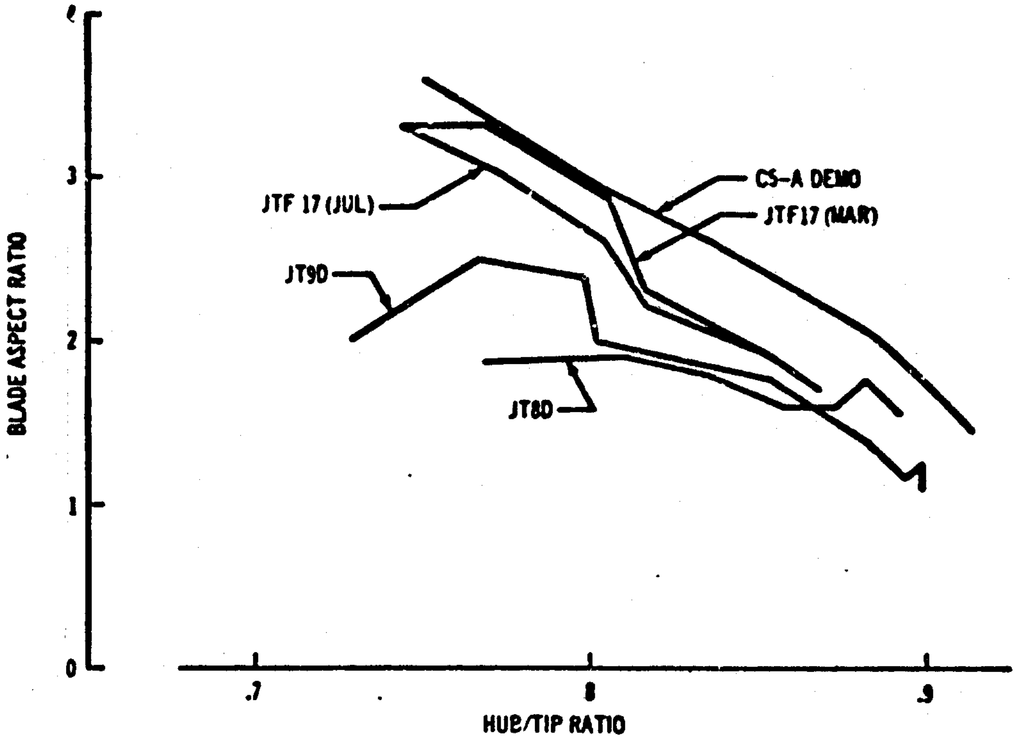

Figure A-19. Aspect Ratio Versus Hub-Tip Ratio for Pratt and Whitney Engines

Combustion efficiency values of 99 percent have been measured in the JT4 demonstrator engine with the ram induction burner. Combustion efficiency can be correlated with the parameter

$$\theta = \frac{P_{in}^{1.75} \times e^{(T_{in}/540)} \times A_{ref}^{0.75} \times D_{ref}}{W_A}$$

as shown in Fig. A-20 for developed annular combustors. It can be further shown that lean blow-out occurs at the value of θ where the curve becomes vertical. Figure A-20 shows values of θ for the P&WA main combustor for various flight conditions. The value of θ at idle conditions appears marginal; however, it is not known to what extent this curve is applicable to the ram induction burner.

The design goal of dmax, an indicator of maximum temperature profile quality of the burner, is stated to be 10 percent. A 120-degree sector rig test has demonstrated this value. However, values of 28 percent to 29 percent have been shown in most of the data available at this time, including engine burner data from the JT4 and JTF17 demonstrator.

Aerodynamic radial distortion testing has been conducted with good distortion attenuation in the 120-degree sector rig test facility. Circumferential distortion testing has not been mentioned to date.

Figure A-21 presents typical operating values of reference Mach number and pressure loss for annular combustors. The design values at SLTO and Mach 2.7 cruise are shown on the figure for the P&WA burner. The figure shows that the design is conservative with regard to pressure loss.

Maximum liner metal temperatures of between 1,800 and 1,850°F in the region near the swirl

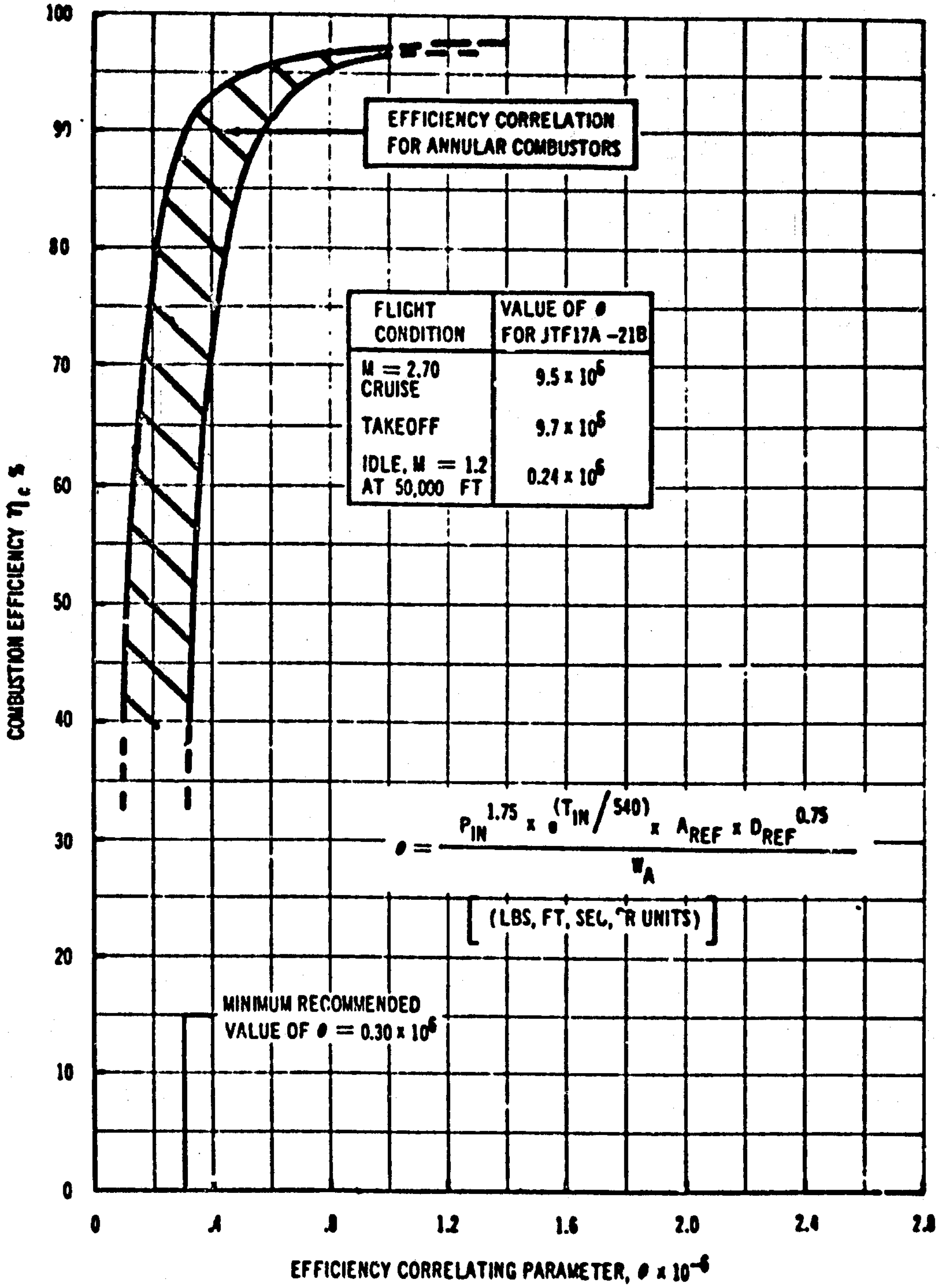

Figure A-20. P&WA JTF17A-21B Combustion Efficiency Correlation

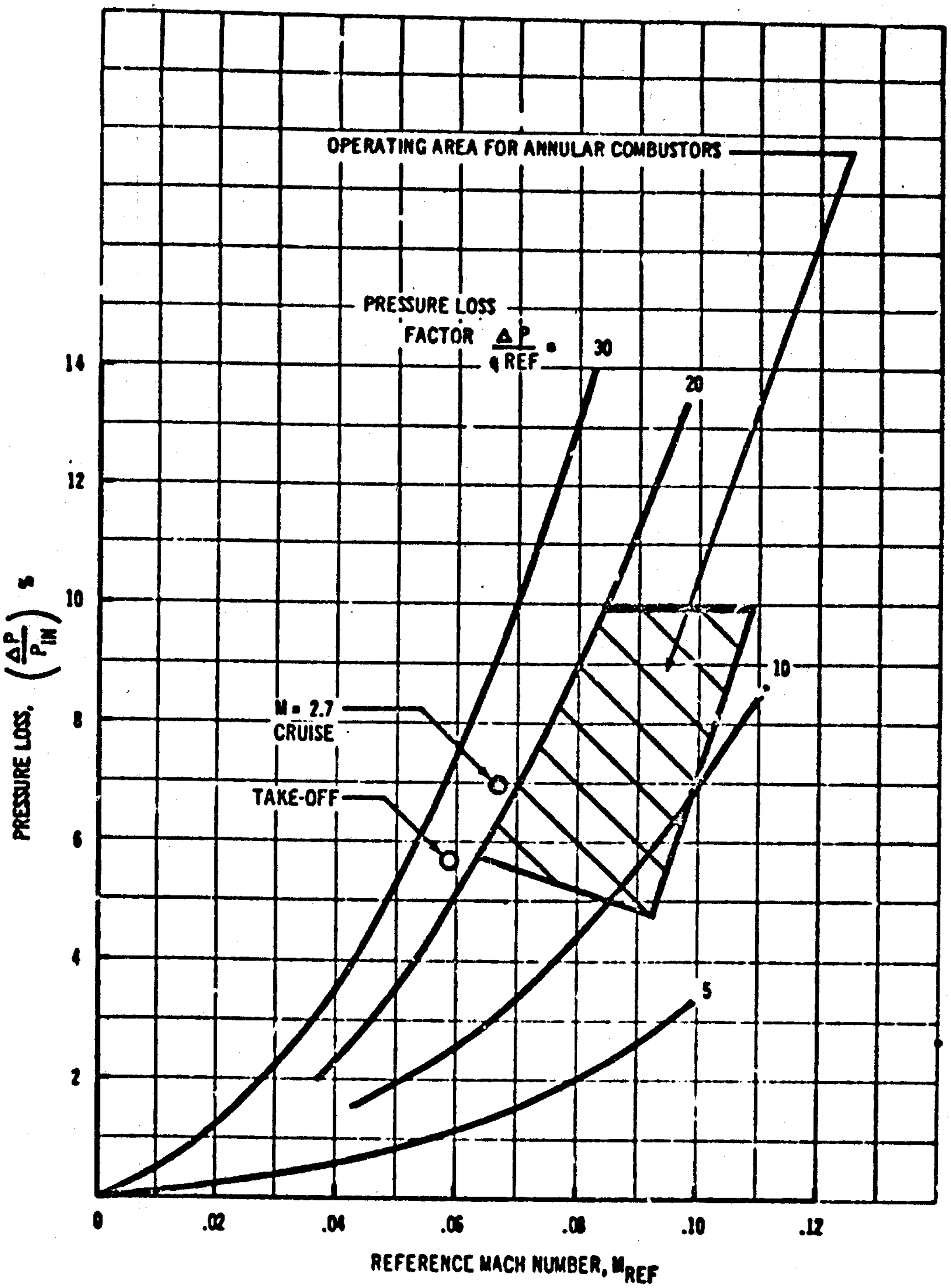

Figure A-21. P&WA JTF17A-21B Burner Pressure Loss

injectors, have been measured. The temperature range along the liner is 1,000 to 1,700°F at takeoff and 1,350 to 1,800°F at cruise. The maximum liner temperature appears high but no information is available on the effect of these temperatures on liner life.

In summary, the ram induction concept is new, and no flight experience is available as a technological base. Combustor efficiency and space heat release rates appear achievable, but the exit temperature profiles require further development. The test program and demonstrated performance to date are encouraging. Overall, the P&WA main burner is not considered to be in one of the major development risk categories.

A3.3 TURBINE

A3.3.1 Aerodynamic Performance

The JTF17 engine employs a three-stage turbine with a controlled-vortex radial aerodynamic loading distribution. The first stage drives the high-pressure compressor, and the second and third stages drive the fan. The turbine aerodynamic design is summarized in Table A-F, where variables which approach severe design values have been circled for emphasis.

This turbine is a highly loaded aerodynamic design in that high cascade Mach numbers and turning angles are generally employed. The low values of rotor exit hub/tip ratio for the second and third stages also indicate a difficult design because of the large radial variation in tangential blade speed.

Because the three stages of this turbine are highly loaded, there is little margin for redesign to extract more work, without an additional turbine stage, should a problem during engine development establish the need for more work extraction.

The values of overall efficiency quoted by P&WA for the high pressure and low pressure stages are compared to other turbines on Fig. A-22 as a function of wheel speed loading parameter, which is a form of Parson's number. The base values of efficiency given by P&WA are seen to be in line with values for other turbine designs.

Also shown on Fig. A-22 are the adjustments which P&WA used to get from the base values to the design values. The adjustment of +2.0 percent for the controlled vortex design as opposed to a free vortex design seems high in light of the highly loaded aerodynamic design of this turbine. It is felt that the +2.0 percent gain would have been appropriate if lower base efficiency values had been chosen (lying closer to the dotted lower limit line for low hub/tip ratio designs).

A second -2.0 percent efficiency adjustment was taken by P&WA for reduced tip clearance and hence is justifiable on an aerodynamic basis. However, tip clearances of 0.02 to 0.035 in. could impose mechanical design problems. As seen on the bottom line of Table A-F, these clearances correspond to less than one-half of one percent of the rotor tip diameter. Development problems may be anticipated in achieving and maintaining these values in operational engines in an SST environment.

A3.3.2 Turbine Life

The JTF17 turbine employs coated P&WA 658 blades with an ultimate life with repair of 10,000 hours. The discs are burst limited and therefore have a long, low-cycle fatigue life. The blades are creep limited.

Care must be taken in the fabrication of forged Astroloy discs to avoid low transverse ductility. P&WA has conducted a large development program with disc suppliers during the past several years and has been able to achieve excellent transverse ductility. P&WA is not expected to have any problems in this area.

The P&WA 658 turbine blades require a coating for oxidation and sulphidation resistance. P&WA has devoted much research and development effort toward effective coatings. It is therefore anticipated that they will be able to develop an adequate coating for the blades.

Boeing has not attempted to judge the low cycle fatigue (LCF) characteristics of the JTF17 turbine because of a lack of sufficient information at this time. No problems are anticipated from internal clogging of the cooling holes, and no additional susceptibility to foreign object damage is expected because of the blade cooling design.

A simplified creep life calculation was made for the JTF17 turbine to estimate the risk in the P&WA-quoted turbine blade creep life. The midspan section of the first rotor was chosen as a point indicative of minimum creep life. Values

Table A-F. Summary of Turbine Aerodynamic Design

Design Quantity	Mild Design	Severe Design	JTF17A (Controlled Vortex) First Stage	JTF17A (Controlled Vortex) Second Stage	JTF17A (Controlled Vortex) Third Stage
At Mean Radius					
Stator exit angle β_2	30°	20°	21.87	34.03	36.18
Stator exit mach $(c/a^*)_3$	0.8	1.15	0.94	0.79	0.76
Rotor inlet relative mach $(W/a^*)_3$	0.4	0.7	0.51	0.48	0.47
Rotor exit relative mach $(W/a^*)_6$	0.8	1.15	0.92	0.8	0.77
Rotor exit axial mach	0.4	0.7	0.43	0.445	0.49
Stator aspect ratio			1.91	3.1	5.65
Rotor aspect ratio			3.33	5.0	
Stator solidity (width/pitch)			1.15	1.45	1.40
Rotor solidity			1.33	1.25	1.24
Rotor exit DHUB/DTIP	0.8	0.5	0.74	0.68	0.48
At Blade Root					
Stator exit angle β_2	30°	20°	26.2	35.55	39.75
Stator exit mach $(c/a^*)_3$	0.8	1.15	1.07	0.90	0.95
Rotor inlet relative mach $(W/a^*)_3$	0.4	0.7	0.72	0.67	0.77
Rotor exit relative mach $(W/a^*)_6$	0.8	1.15	0.95	0.89	0.93
Rotor turning angle	80°	120°	107.5	94.7	84.7
Degree of reaction (W6/W3-1)	0.35	0	0.31	0.26	0.?
Other Design Variables			HP	LP = Stgs. 2 & 3	
Stage work, Δh			112.38	113.72	
Stage efficiency			86.9	88.0	
Mean blade speed, Um			1144.	782.	
Loading parameters, $U^2/2gJ\Delta h$			0.232	0.215	
Tip Clearance			0.020"	0.030"	0.035"
Tip clearance, % blade height			0.345	0.353	0.310
Tip clearance, % tip diameter			0.055	0.0075	0.008

Blocked variables approach severe design

of temperatures and pressures were obtained from the JTF17 proposal.

Several values used in the calculations are summarized below:

- o Calculated tensile stress = 19,200 psi
- o Design factor = 1.0
- o Design stress = 19,200 psi
- o Average rotor gas temperature = 2,010°F
- o Average metal temperature = 1,640°F
- o Cooling effectiveness = 0.421
- o Material = P&WA 658
- o Larson-Miller factor (for P&WA 658 to 1.0 percent creep) = 47,800
- o Estimated creep life = 3,160 hours

The calculated life is about one-third of the life quoted by P&WA, but this is not too unreasonable

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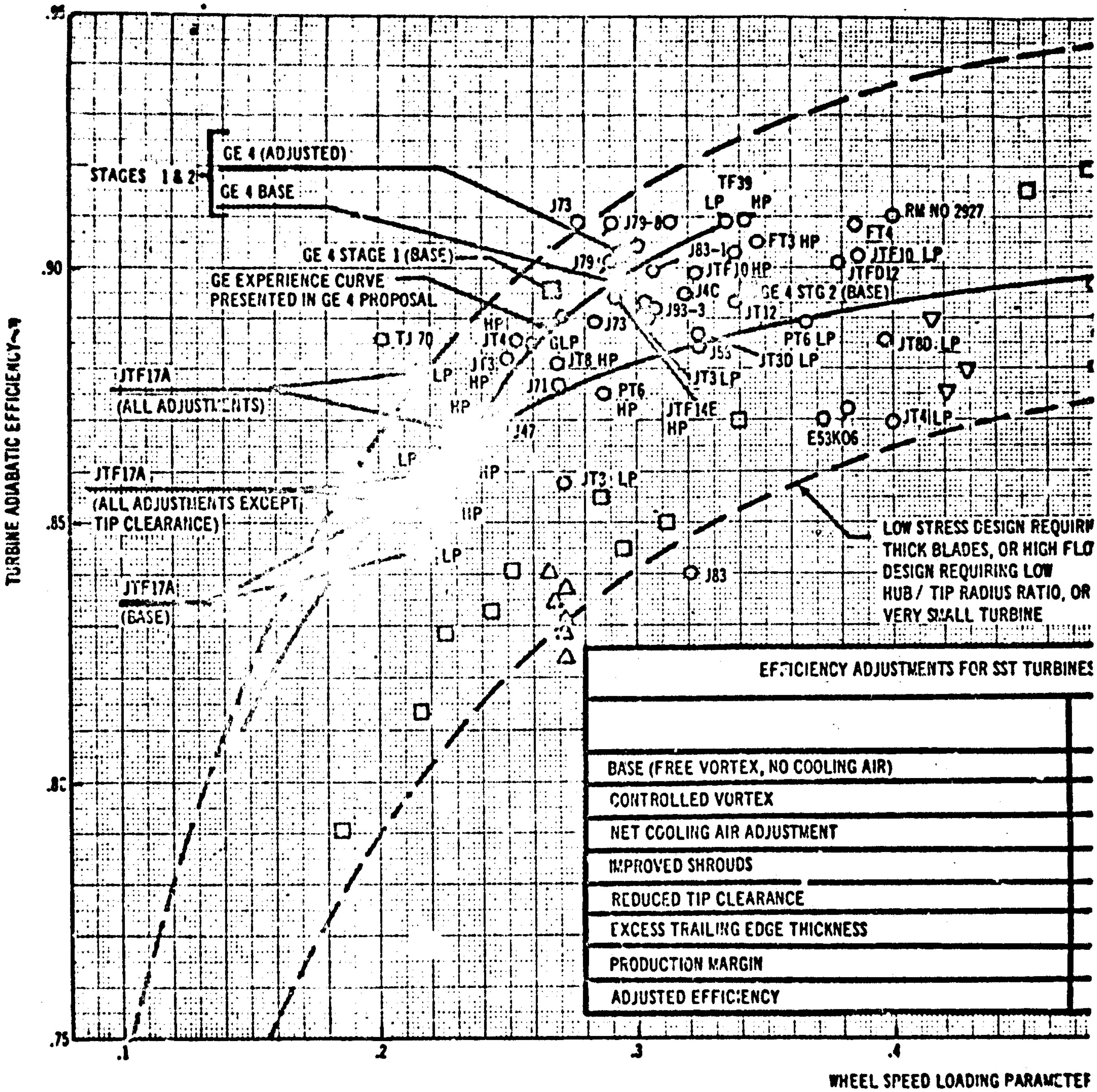

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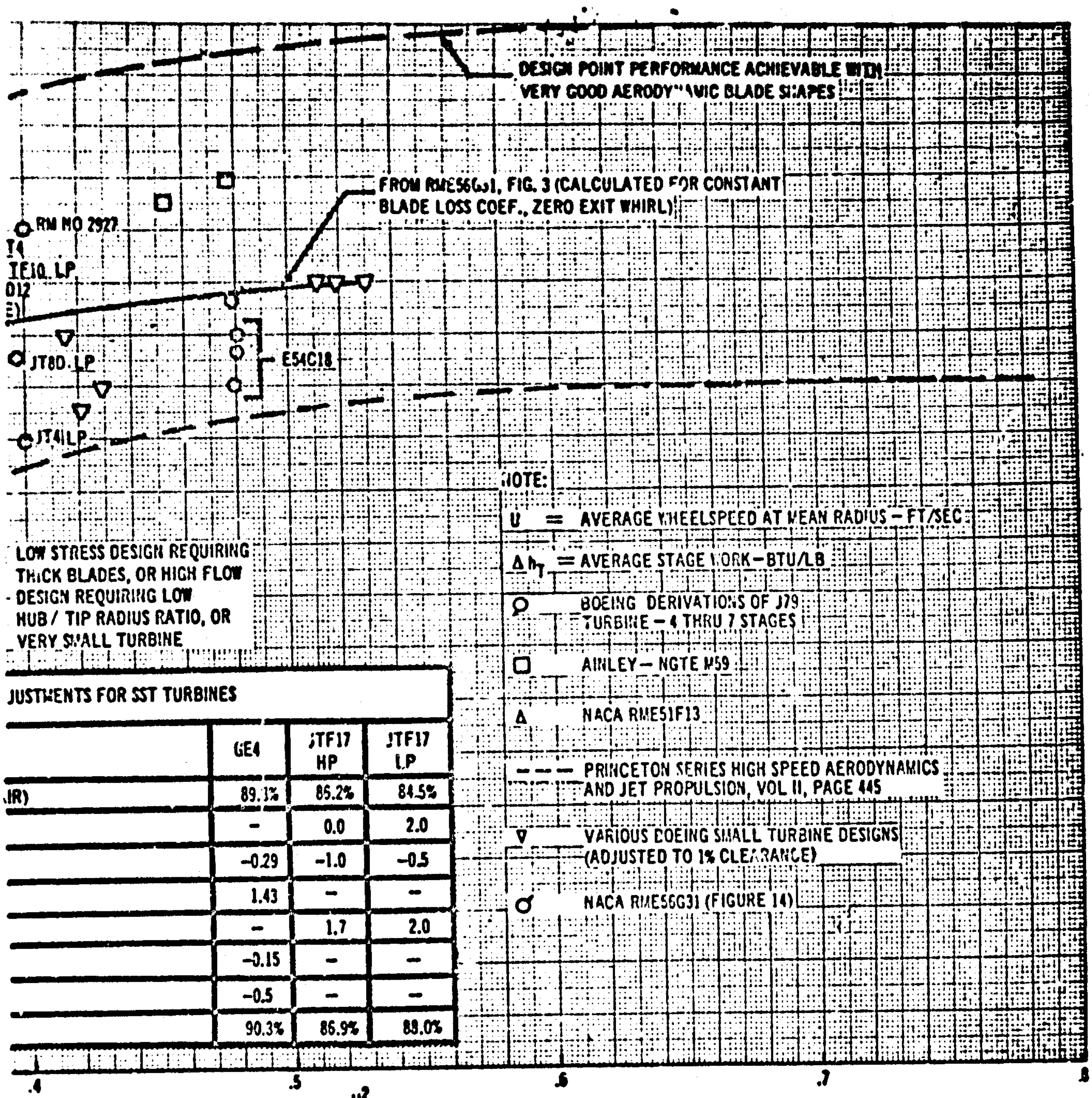

JUSTMENTS FOR SST TURBINES

	GE4	JTF17 HP	JTF17 LP
IR)	89.1%	85.2%	84.5%
	–	0.0	2.0
	–0.29	–1.0	–0.5
	1.43	–	–
	–	1.7	2.0
	–0.15	–	–
	–0.5	–	–
	90.3%	86.9%	88.0%

Figure A-22. Correlation of Turbine Efficiency with Wheelspeed Loading Parameter

2

considering that a small change in the Larson-Miller factor or the blade temperature will cause a large change in the calculated life.

The estimated creep life is based on a P&WA design criterion of 1 percent creep. This criterion would make little difference in the life calculation for very high stress values (greater than 50,000 psi); however, it has a large effect on the calculated life for stresses in the 20,000-psi range.

In summary, the JTF17 turbine is a highly loaded aerodynamic design with high cascade Mach numbers and low exit hub/tip ratios. It appears that the goal efficiency will be difficult to achieve. The rotor tip clearances of 0.02 to 0.035 in. are required to achieve the stated efficiencies. These clearances will be difficult to maintain in operational engines. An increase to a more conventional 0.08-in. clearance will cost 2 percent in turbine efficiency. The JTF17 turbine aerodynamic performance is classified as a Category 2 risk item.

Typically a fan engine turbine blade has rather high stresses at mid-span. In this engine, the average metal temperature of 1,640°F, together with the material selected for the turbine blades (P&WA 658) and the high stresses involved, will make it difficult to achieve the creep life of 10,000 hours. The JTF17 turbine life is classified as a Category 1 risk item.

A3.4 AUGMENTOR

The thrust augmentor for the Pratt and Whitney engine is a duct heater, the first to be proposed for flight application. The heat release rate is substantial as can be seen in Fig. A-2. Because of the similarity of design, the demonstrated heat release rate of the primary burner tends to substantiate the duct burner design g[illegible].

The goal thrust-averaged combustion efficiency is 97 percent over the augmentation range from minimum duct heat to 2,500°F gas temperature, with a decrease in efficiency to 90 percent at 3,600°F gas temperature. According to P&WA, this implies a chemical combustion efficiency of nearly 100 percent over a broad range of augmentation temperature. Test results shown in Fig. A-23 do not verify these goals, especially in the cruise range of fuel-air ratios. In this region (fuel-air ratio = 0.014 to 0.022), only one cruise-demonstrated efficiency test point is shown, at around 96 percent. Other points near this fuel-air spectrum are below the goal efficiency and show a wide range of scatter.

Figure A-3 may be used as an indicator of expected efficiency. The relatively low duct heater entry temperature is offset by low reference velocity. Therefore, based on this trend curve, high values of efficiency should be expected for the duct heater.

Aerodynamically, the duct burner must be designed to accommodate the flow exiting from the fan. The specified goal cold flow and hot flow losses have been demonstrated with rig tests. Considerations should be given to pressure losses in the presence of fan distortion into the diffuser and burner as they will occur in actual practice. P&WA has stated that rig test work has been completed with simulated steady-state distortion, and that the diffuser and burner handled it well. No data from these tests have been made available at this time. In addition, unsteady distortion, (turbulence) testing has not been reported for this burner.

Burner liner temperatures during cruise are estimated to average between 1,200 and 1,400°F. Maximum temperatures are reported to be 1,550 and 1,440°F for the outer and inner liners respectively, and with proper design control, long life should be attained at these temperatures. Thus, liner life is not considered in a major risk category.

In summary, the burner design concept is new, and is without previous supersonic flight experience as a technology base. The augmentor must perform under a wide range of flow conditions from the fan and at high combustion efficiency levels. Because of the effect of duct burner efficiency on cruise SFC, the duct burner is classified as a Category 2 development risk item. These comments are based on steady flow operation. The augmentor dynamics are covered in Section A3.7.

A3.5 EXHAUST NOZZLE/REVERSER

A3.5.1 Exhaust Nozzle

In its August report to Boeing (P&WA FR-66-100), P&WA supplied data points from tests of the 650 lb/sec JTF17A-20B nozzle. Test data for takeoff, subsonic cruise, Mach 1.2 climb, and supersonic

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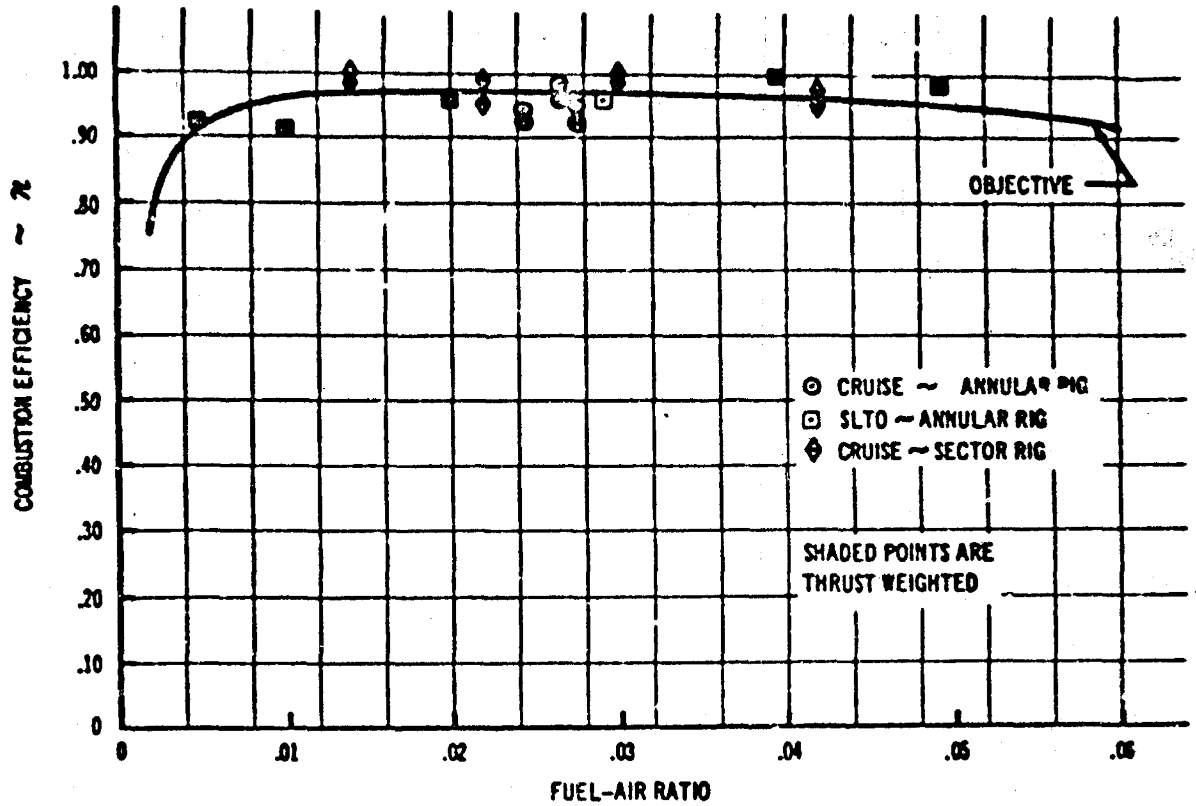

Figure A-23. Duct Heater Combustion Efficiency

cruise are shown in Fig. A-24 through A-27. On each figure, thrust coefficient goals are shown at the operating pressure ratios for both the -20 and -21B engines. These goals are derived from the P&WA performance decks. At the subsonic and transonic conditions, the P&WA model performance does not meet the -20B goals. In addition, the -21B goal is higher than the -20B goal at Mach 1.2 which makes the performance decrement even greater at that condition. The specified performance is met at supersonic cruise, the most important condition, but further nozzle development will be required to improve the off-design thrust coefficients.

A3.5.2 Reverser

With reverser clamshells in the reverse position, a considerable gap exists between the clamshells and the shroud. The exhaust gas escaping rearward causes a double penalty to potential reverse thrust. In order to achieve the reverse thrust goal, the exhaust gas must be directed forward within 10 to 30 degrees of the engine centerline through the blow-in door opening. Depending upon the design blow-in door angle, nacelle flow attachment, with attendant reingestion problems, is possible. P&WA model test performance demonstrates that the reverse thrust goal of 40 percent can be achieved.

In summary, although P&WA tests tend to substantiate the nozzle performance level at supersonic cruise, further development is required to achieve the nozzle performance goals at other flight conditions. In view of the strong effects of nozzle performance on airplane performance, the nozzle is considered a Category 2 development risk item until specific tests prove that the performance levels can be attained.

Pratt and Whitney Aircraft reverser model tests indicate that the reverse thrust design goals will be met. However, control of reverse gas flow direction and distribution is expected to present

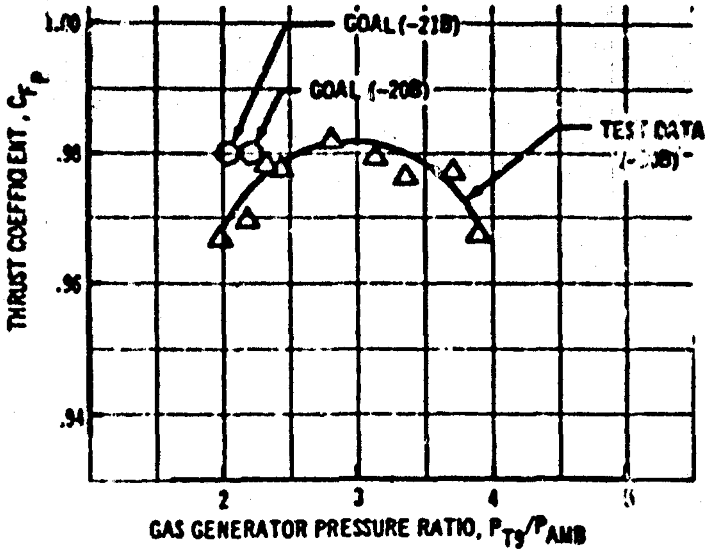

Figure A-24. P&WA Nozzle Test Data Takeoff

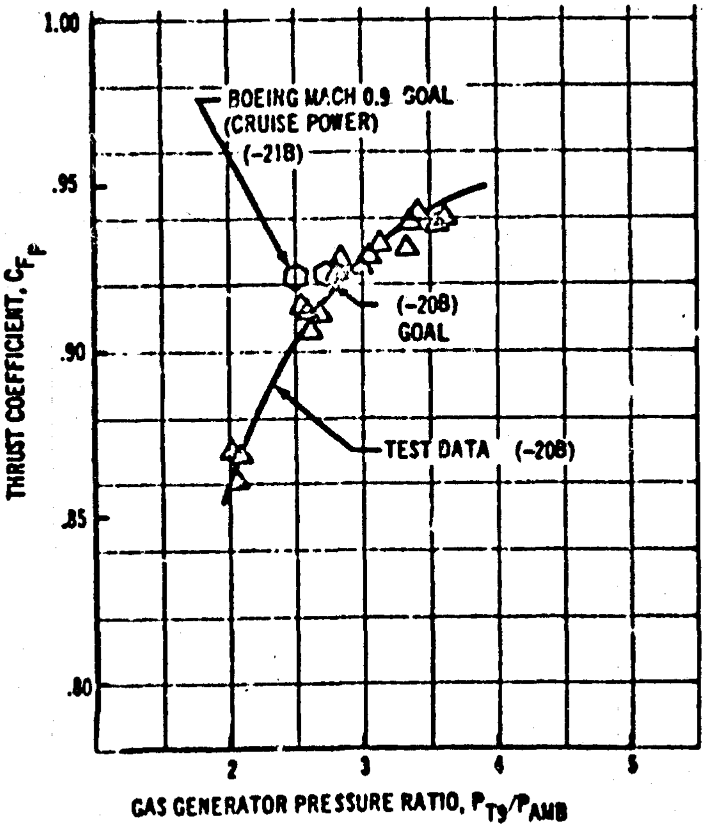

Figure A-25. P&WA Nozzle Test Data Mach .9 Cruise

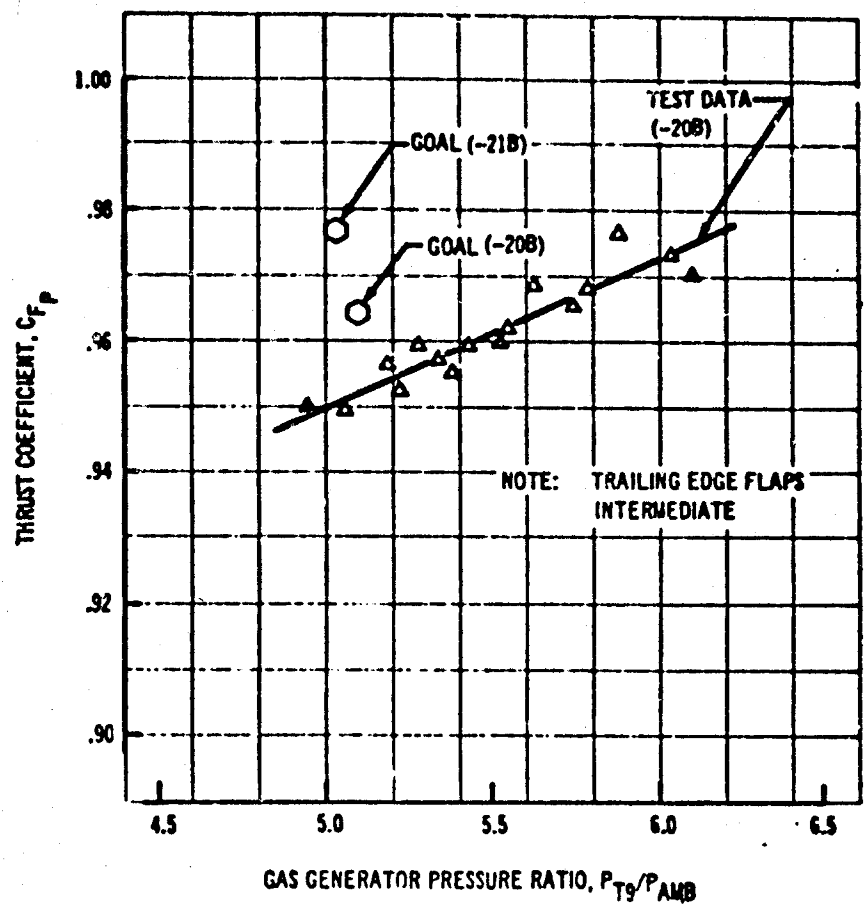

Figure A-26. P&WA Nozzle Test Data Mach 1.2 Climb

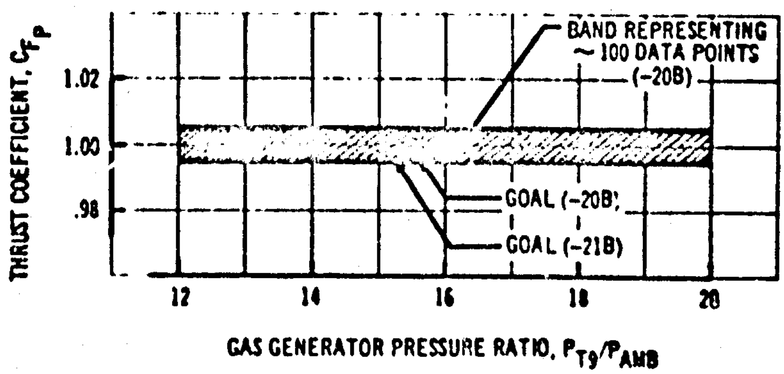

Figure A-27. P&WA Nozzle Test Data Mach 2.7 Cruise

problems with the present design, in which the flow must exit through the blow-in doors. The flow angle is such that the flow could cling to the engine nacelle and enter the engine inlet. For this reason, the reverser concept is considered to be a Category 2 risk item.

A3.6 ENGINE WEIGHT

Boeing does not feel qualified to make a detailed weight analysis of the JTF-17 engine. Nevertheless, two comments are in order.

There exists no precedent for the offered nozzle-thrust reverser system, and gas flow requirements to avoid engine ingestion and impingement on airplane surfaces could very well lead to a weight increase.

There is reason to expect a weight increase in the engine fan/compressor as a part of the effort to achieve the required steady-state distortion and dynamic distortion tolerance. These factors classify the weight of the JTF17 as a Category 2 development risk item.

A3.7 CONTROLS AND ENGINE DYNAMICS

A3.7.1 Control System

A functional description of the control system is presented in the following paragraphs.

A3.7.1.1 Main Fuel Control (See Fig. A-28)

a. Primary functions

Control engine speed during steady-state and transient operation of the engine.

Control fuel input during transient operation to prevent compressor stall, engine flame-out, or turbine overtemperature.

Position the variable IGV of the HP compressor.

b. System operation

Steady-state

Fuel flow is varied to maintain gas generator rpm (N_2) as scheduled by thrust lever angle (TLA) and compressor inlet total temperature (T_{T2}). An approximate value of engine fuel flow/burner pressure ratio (W_F/P_B) is scheduled as a function of TLA and T_{T2}. Scheduled rpm is then maintained by biasing the above value of W_F/P_B with a W_F/P_B proportional to the difference between scheduled and actual rpm. Actual fuel flow to the engine is determined by the product of required W_F/P_B and sensed burner pressure (P_B).

At low corrected speed, a steep droop slope is used; at high corrected rpm, much more droop is allowed.

The HP compressor IGV is positioned as a function of N_2 and T_{T2} (only 2 positions).

Transients

Fuel flow is limited by acceleration and deceleration schedules to provide rapid rpm change without encountering compressor stall, engine flame-out, or exceeding engine temperature limits. These schedules are a function of P_B, T_{T2} and N_2.

c. Sensed quantities

TLA

N_2

T_{T2}

P_B

d. Controlled outputs

Gas generator fuel flow (W_F)

HP rotor IGV angle

A3.7.1.2 Duct Heater Fuel Control (See Fig. A-29)

a. Primary function

Schedule fuel to the augmentor during steady-state and transient operation of the engine in the augmented range.

b. **System operation**

Steady-state

Duct heater fuel flow-burner pressure ratio (W_{FR}/P_B) is scheduled as a function of thrust lever angle (TLA) and compressor inlet total temperature (T_{T2}) to provide the desired temperature rise and thrust augmentation in the duct heater: Actual fuel flow to the duct heater is determined by the product of W_{FR}/P_B and sensed burner pressure (P_B).

Transient operation

Two conditions must be met before augmentation will be initiated:

The power lever must be in augmentation range and engine rpm must exceed 80 percent.

If augmentation initiation does not produce an A_{8F} change within one second, then augmentation initiation is terminated automatically. The power lever must be recycled to Maximum Dry before a new attempt can be made to light.

Augmentation initiation includes an automatic preopening of A_{8F}. This is removed after the light-off is completed.

Following light-off, augmentor fuel flow is increased until the desired level is reached.

c. **Sensed quantities**

TLA

Compressor discharge pressure (P_B)

T_{T_2}

d. **Controlled outputs**

Augmentor fuel flow

A3.7.1.3 **Duct Nozzle Area Control (See Fig. A-30)**

a. **Primary function**

Control duct nozzle exhaust area during steady state and transient operation of the engine.

b. **System operation**

Steady state

During dry operation, duct nozzle area (A_{8F}) is scheduled by thrust lever angle (TLA) and compressor inlet total temperature (T_{T2}) from full open at Idle power to nearly full closed at Maximum Dry power.

During augmented operation, the nozzle is controlled to maintain corrected engine airflow as a function of compressor inlet total temperature. An approximate value of nozzle area is scheduled as a function of TLA and T_{T2}. Duct Mach number just aft of the fan is determined by sensing total and static pressures (P_{T3} and P_{S3}) and combining these pressures in the ratio

$$\frac{P_{T3} - P_{S3}}{P_{T3}} = \frac{\Delta P}{P}.$$

The $\Delta P/P$ required to maintain constant corrected airflow is scheduled as a function of gas generator rpm (N_2) and compressor face total temperature (T_{T2}). This value of $\Delta P/P$ is maintained by biasing the duct nozzle area through a closed-loop control.

Transient

Several features are included in the duct nozzle control to improve engine transient response and augmentor light-off. These include:

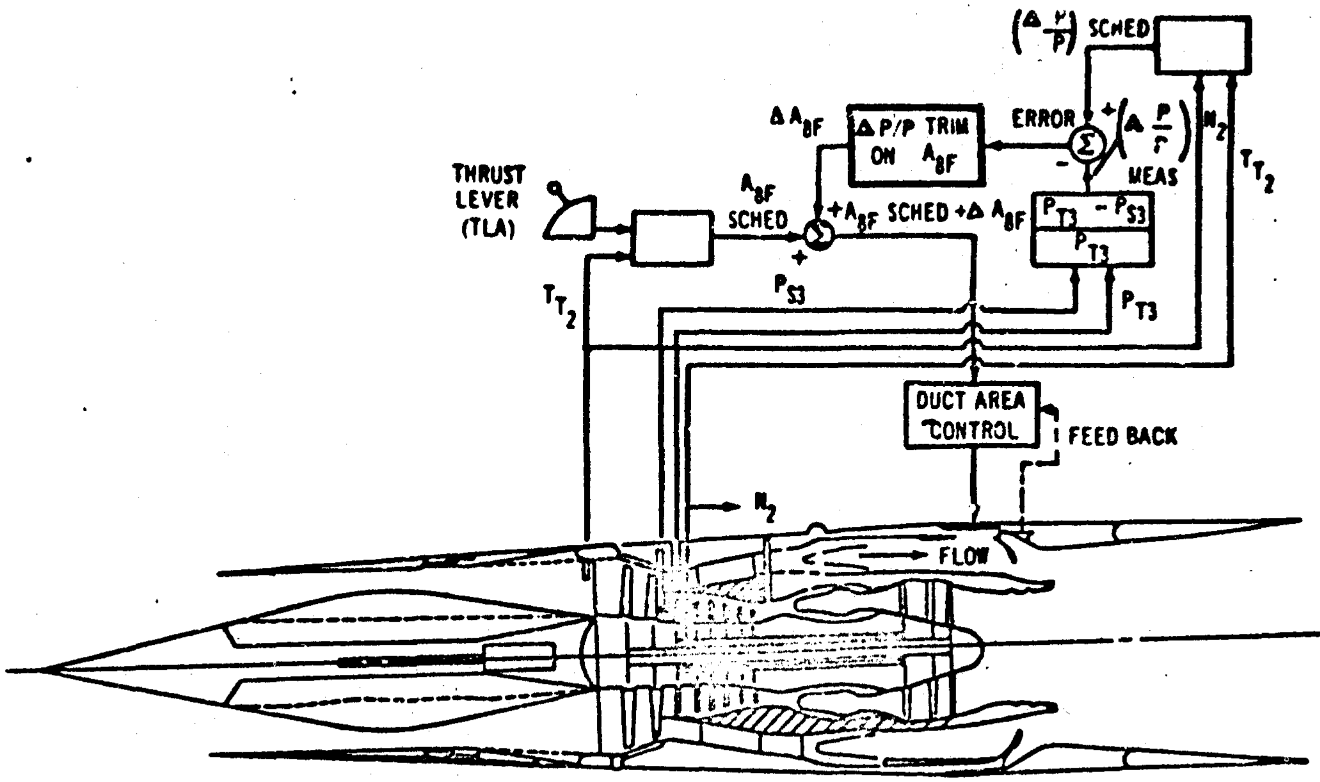

Figure A-30. Schematic of Duct Nozzle Area Control System

$$A_{8F} = f(N_2, T_{T_2}, TLA, \frac{P_{T3} - P_{S3}}{P_{T3}})$$

(1) An increase of the nozzle area one sec before augmentor ignition to increase inlet stability margin.

(2) A signal to the duct heater fuel control to cut off augmentor fuel flow when the measured Δ P/P corresponds to a corrected airflow more than 4 percent above the scheduled value.

(3) A signal to the main fuel control to decrease gas generator fuel flow (W_F) when the measured Δ P/P corresponds to a corrected airflow more than 4 percent above the scheduled value. This will prevent excessive rotor overspeed following duct heater flame-out.

c. Sensed quantities

Thrust lever angle (TLA)

Fan discharge total pressure (P_{T3})

Fan discharge static pressure (P_{S3})

Gas generator rpm (N_2)

Compressor inlet total temperature (T_{T2})

Nozzle area (position feedback)

d. Controlled output

Duct nozzle area (A_{8F})

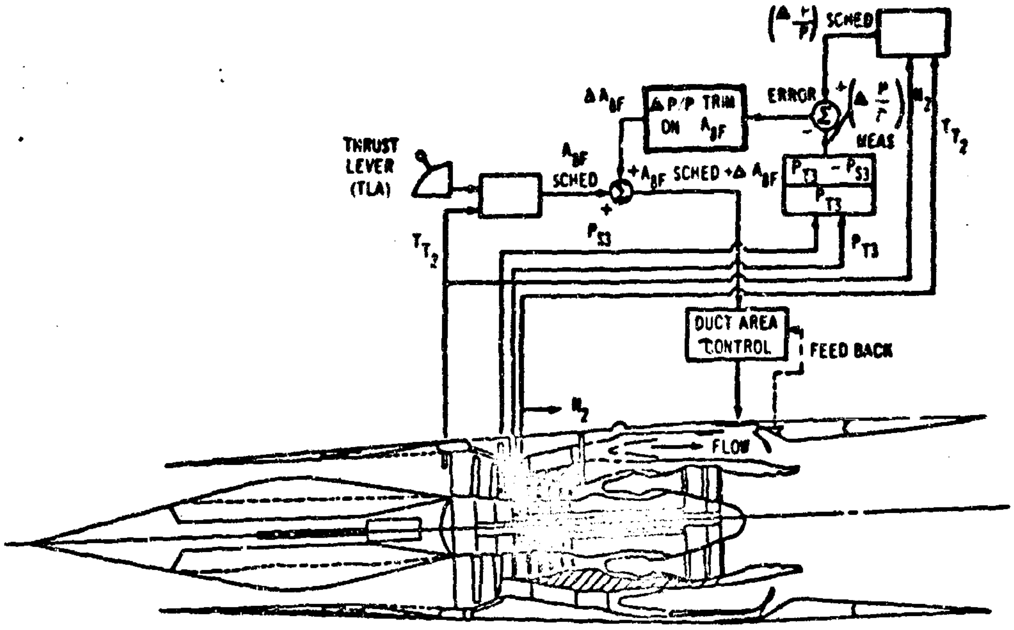

Figure A-30. Schematic of Duct Nozzle Area Control System

$$A_{8F} = KN_2, T_{T_2}, TLA, \frac{P_{T3} - P_{S3}}{P_{T3}}$$

(1) An increase of the nozzle area one sec before augmentor ignition to increase inlet stability margin.

(2) A signal to the duct heater fuel control to cut off augmentor fuel flow when the measured Δ P/P corresponds to a corrected airflow more than 4 percent above the scheduled value.

(3) A signal to the main fuel control to decrease gas generator fuel flow (W_F) when the measured Δ P/P corresponds to a corrected airflow more than 4 percent above the scheduled value. This will prevent excessive rotor overspeed following duct heater flame-out.

c. Sensed quantities

Thrust lever angle (TLA)

Fan discharge total pressure (P_{T3})

Fan discharge static pressure (P_{S3})

Gas generator rpm (N_2)

Compressor inlet total temperature (T_{T2})

Nozzle area (position feedback)

d. Controlled output

Duct nozzle area (A_{8F})

A3.7.1.4 Background
Experience gained with previous P&WA engine designs is directly applicable to the main fuel control and the duct heater fuel control of the JTF17.

A3.7.1.5 Special Features
A fundamental aspect of the JTF17 engine is the fact that the primary nozzle is fixed and the secondary nozzle variable. Changes in the variable nozzle area directly affect fan duct airflow, and hence, total engine airflow, which is an advantage of the fan cycle. However, such nozzle area changes have little effect on gas generator control parameters such as HP spool rpm (N_2) and T_{T4} (turbine inlet temperature). Turbine inlet temperature control through the main fuel control is apparently not sufficiently accurate. P&WA had added an extra T_{T4} trim to maintain T_{T4} within desired limits.

The variable fan duct nozzle is used to control fan duct flow during augmented operation. This particular system incorporates two subsystems, each of which has a high response: the hydraulic nozzle positioning system and the fan duct Mach number sensor and computer.

The former needs to have rapid response, while the latter is a link in compensating for duct flow transients. This situation could lead to instability of the overall nozzle control system. P&WA acknowledges this under "anticipated problem areas," and states that extensive dynamic bench and engine testing will be conducted to obtain satisfactory system performance.

In case of control problems, the P&WA design offers the opportunity to include the variable IGV of the HP rotor in the control system. Because this is essentially a one-stage variable stator, the effect of the change will be small. In the following discussions it should be borne in mind that it would only affect the HP rotor and gas generator airflow, not the duct airflow.

A3.7.1.6 Summary
P&WA is offering a control system which in one important aspect has no direct precedent, namely the control of fan duct airflow by means of variable nozzle area using a pitot static tube as a measure of airflow. The P&WA control system offers only the variable IGV of the HP rotor to smooth out high-frequency control problems. This IGV, however, will have very limited influence on the HP compressor.

A3.7.2 Dynamics

A3.7.2.1 Background
P&WA has in the last year or two started to use more sophisticated digital simulation techniques. Recently Boeing was given a very simple digital mathematic model of the engine and two months ago, a more sophisticated model. This model is still not as well developed as is required for good control system studies. An example is main burner and DH flame-out. The user of the P&WA program must as an input, select whether these burners will or will not flame out during a transient. Another matter which the P&WA program does not take into account is the changing inlet distortion during transients and its effect on fan and compressor maps.

Comparisons between J58 digital analog runs and experimental data were made available and good correlation was obtained. However, no turbofan correlations are available.

Boeing has not been provided complete information about the details of the new P&WA digital simulation of the JTF17 control system.

In summary, P&WA has only recently provided Boeing with a fairly sophisticated digital simulation program. However, this program does lack some important capabilities. The details of the control system simulation are not revealed by P&WA.

A3.7.2.2 Nozzle Area Control
During unaugmented operation, the nozzle area control schedule is only a function of HP spool rpm and T_{T2}. During augmented flow, the duct nozzle area is controlled to keep fan duct Mach number constant for the purpose of keeping engine airflow constant. Fan duct Mach number is measured by a pitot tube. This is a difficult instrumentation problem considering that the Mach number is about 0.5, that the sensor must be capable of responding rapidly, and considering the effect of steady-state and dynamic changes in pressure profiles at the pitot tube (in contrast to pressure changes without profile shifts).

Such changes could originate within the fan itself because of changes in operating conditions, or from internal duct pressure disturbances.

A3.7.2.3 Duct Heater Light-Off
Transients during a duct heater light-off at Mach 2.7 cruise are shown in Fig. A-31. The transients clearly show the effect of opening A_{8F} prior to light-off: the engine airflow increases and the normal shock moves away from the throat. Light-off therefore will not cause unstart, though the transients are significant in their rates and absolute values.

Duct heater light-off at speeds below Mach 2.7 (when the bypass doors are partially open) is a different matter. After opening of A_{8F}, the bypass doors close to bring the shock back to its normal position. This takes place just before light-off. The result is that light-off causes the shock to move forward from its normal position, coming much closer to the throat than in Fig. A-31. Hard light-offs are prevented by the light-off interlock between DH fuel control and A_{8F}.

A3.7.2.4 Duct Heater Blow-Out
Duct heater blow-out transients (Mach 2.7) are shown in Fig. A-32. Blow-out causes a sudden increase in fan airflow, which shuts down the duct heater fuel flow and nozzle A_{8F} as rapidly as possible. The rapid increase in airflow will cause super critical operation of the inlet normal shock, which could cause engine stall. The system is designed so that an airflow transient greater than 4 percent will result in shutdown of duct heater fuel flow. This fuel flow shutdown is essential because an uncontrolled heater relight is unacceptable. Only a controlled relight at minimum fuel/air ratio produces sufficiently small airflow transients. Relight at high fuel/air ratios would produce high airflow changes. Duct heater relight must be initiated by recycling the thrust lever through the Maximum Dry position. The P&WA augmentor operating limits are shown in Fig. A-33.

A3.7.2.5 Inlet Unstart
An inlet unstart simulation (Fig. A-34) shows that the fan surges and the airflow increases above the nominal value. At 4 percent above nominal flow, the DH Mach number computer shuts down the duct heater fuel flow and closes down the A_{8F} as described above. The duct heater relight sequence is described in the preceding paragraph.

A3.7.2.6 Fan and Compressor Surge
P&WA program planning indicates that fan distortion testing will not start until after the SST airframe contractor has been selected. P&WA has not presented a plan for testing with steady-state distortion and turbulence.

The aspect ratios of the HP compressor blades are high. In fact, they are close to those of the C-5A demonstrator HP spool and higher than those of the JT8D and JT9D spools. No test data have been presented.

With respect to the fan, distortion attenuation in the tip region is an unknown. The P&WA proposal points at the well-known flow-work relationship of a rotor to explain that the tip of a rotor blade will attenuate distortion much more than the root. This is based on subsonic flow theory and is probably even true around Mach 1 (which is where the fan tip operates in cruise). However, for subsonic flight and transonic acceleration the fan tip relative Mach number is approximately 1.67 and subsonic theory does not apply. Under these conditions shock and separation phenomena (as indicated by the low efficiency) can create and expose the duct burner and duct flow control sensors to a non steady flow. Pratt and Whitney Aircraft has not revealed a clear cut approach to this problem.

The use of the variable IGV for control purposes is of limited value because only one row of vanes is used. Redesign of fan or compressor to provide more chord (lower aspect ratio), an extra stage, or a fan IGV are possible problem solutions. However, these approaches are particularly unfavorable to a four-bearing engine layout and add both weight and complexity.

In summary, because transients in fan inlet flow will occur as the direct result of the nozzle control oscillations, the HP compressor stall margin for both distorted and fluctuating flow must accept the transients. There are no data available for either the fan or compressor which show that this compressor or fan design tolerate these transient disturbances or distortion. P&WA data show that a duct burner blowout will cause fan stall, mainly because the duct pitot tube and nozzle area control cannot respond fast enough.

A3.7.3 Controls and Dynamics Summary
Because of the unknowns regarding the control system concept, and because the r[illegible]ponse of the engine system to transient disturbances

may, to a certain degree, be fundamental to the turbofan cycle. Boeing believes the control and engine dynamics represent a Category 3 development risk. This is believed to be true until sufficient component and engine testing, together with detailed mathematic model studies, have shown that no fundamental problems exist.

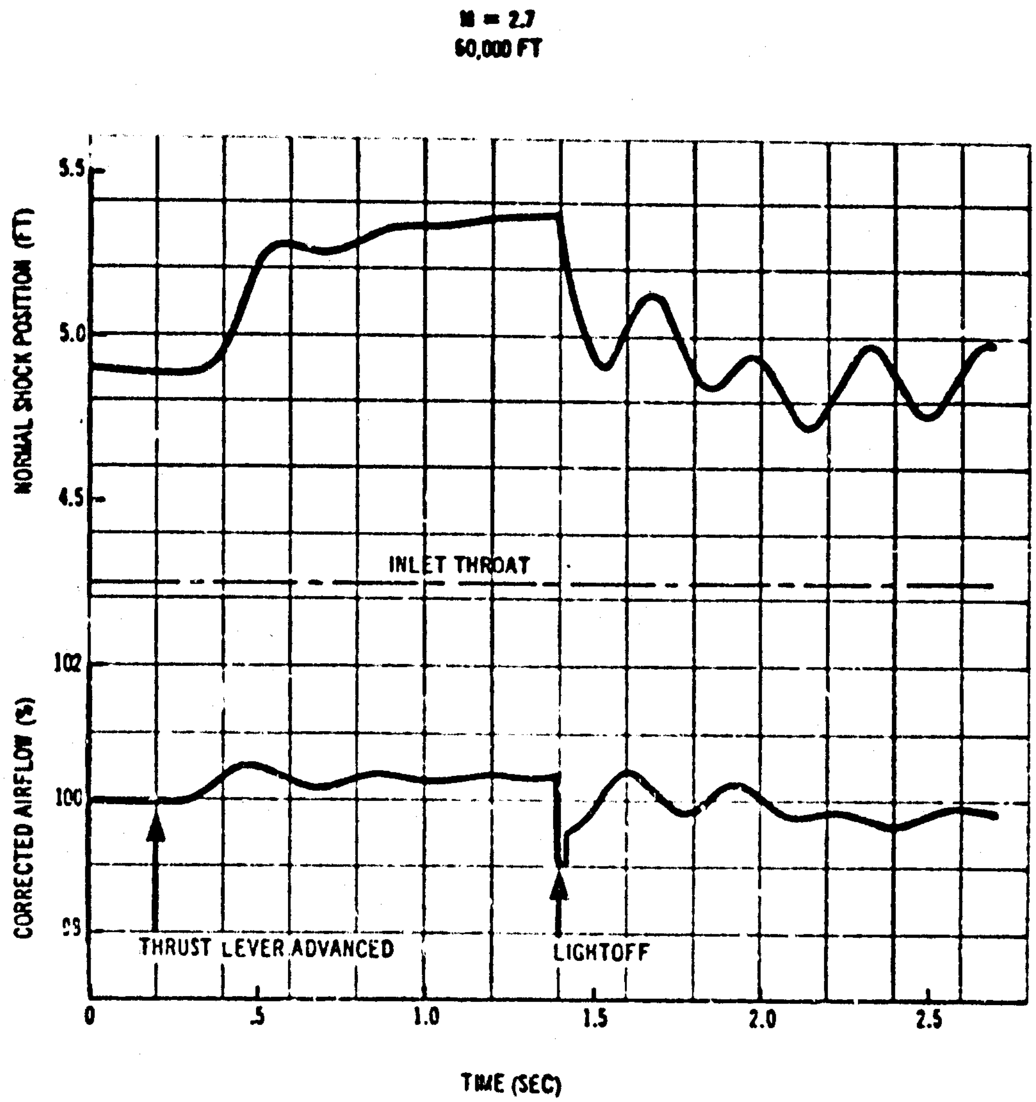

Figure A-31. P&W Augmentor Normal Lightoff (Engine and Inlet)

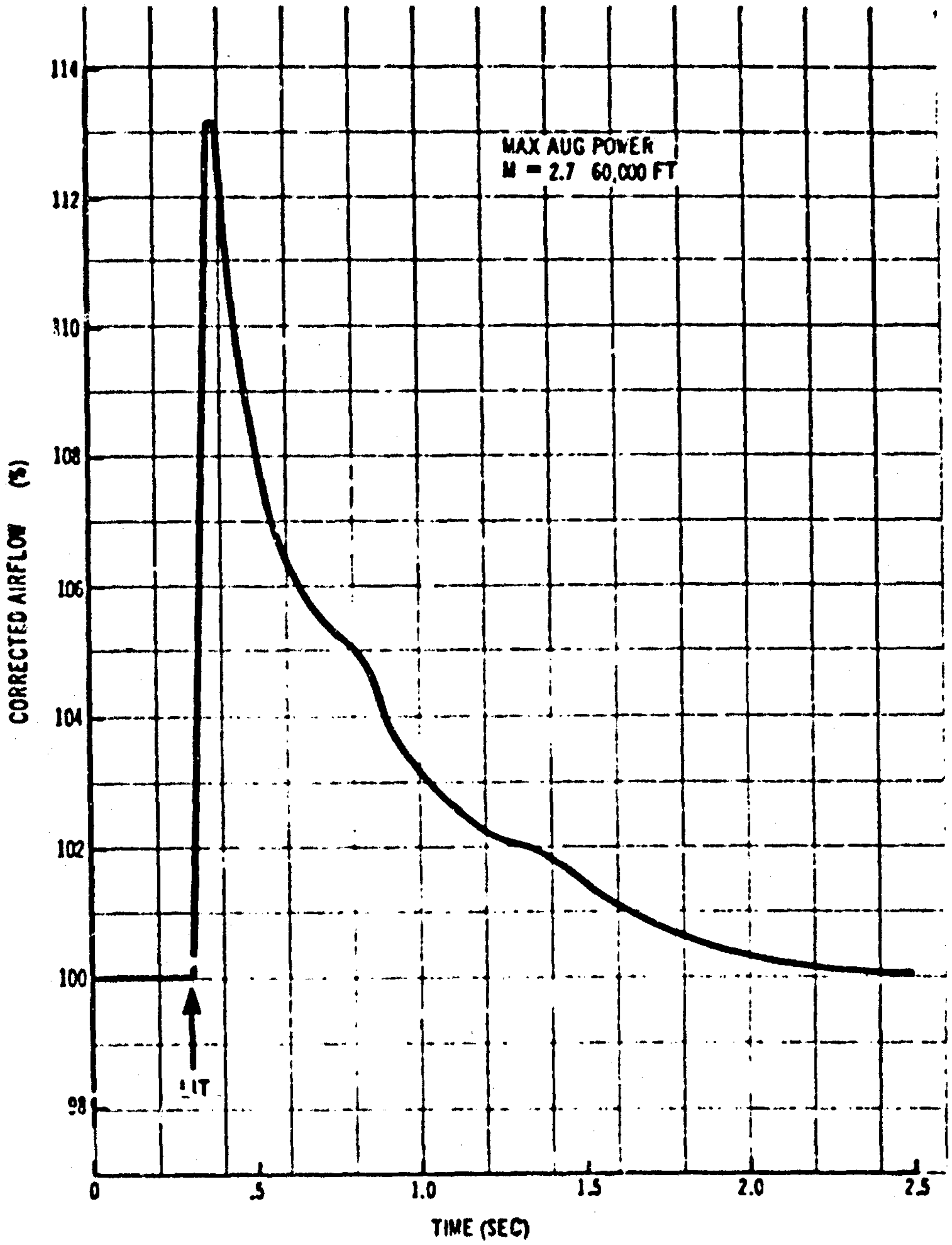

Figure A-32. P&W Augmentor Blowout

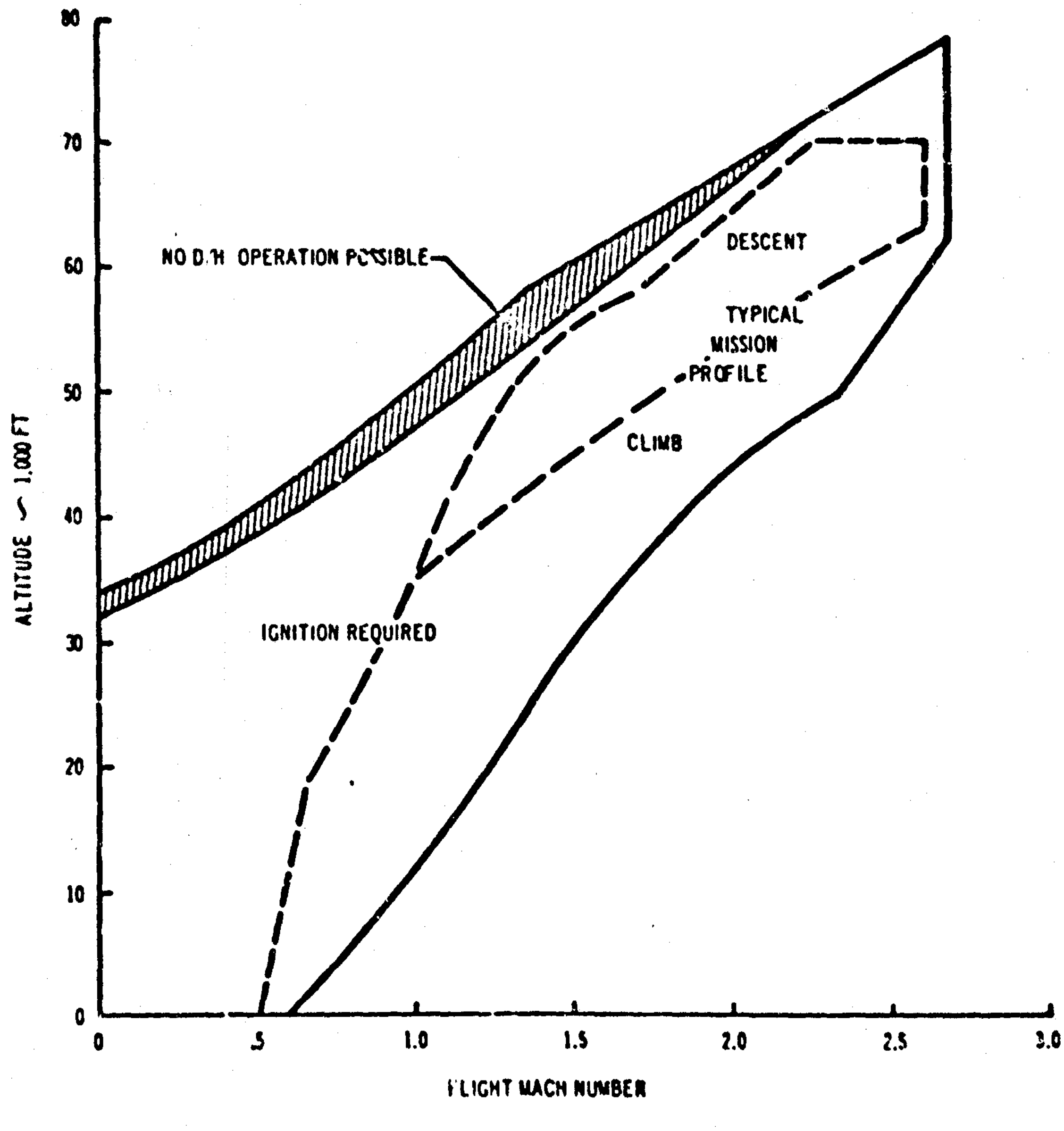

Figure A-33. P&W Augmentor Limits

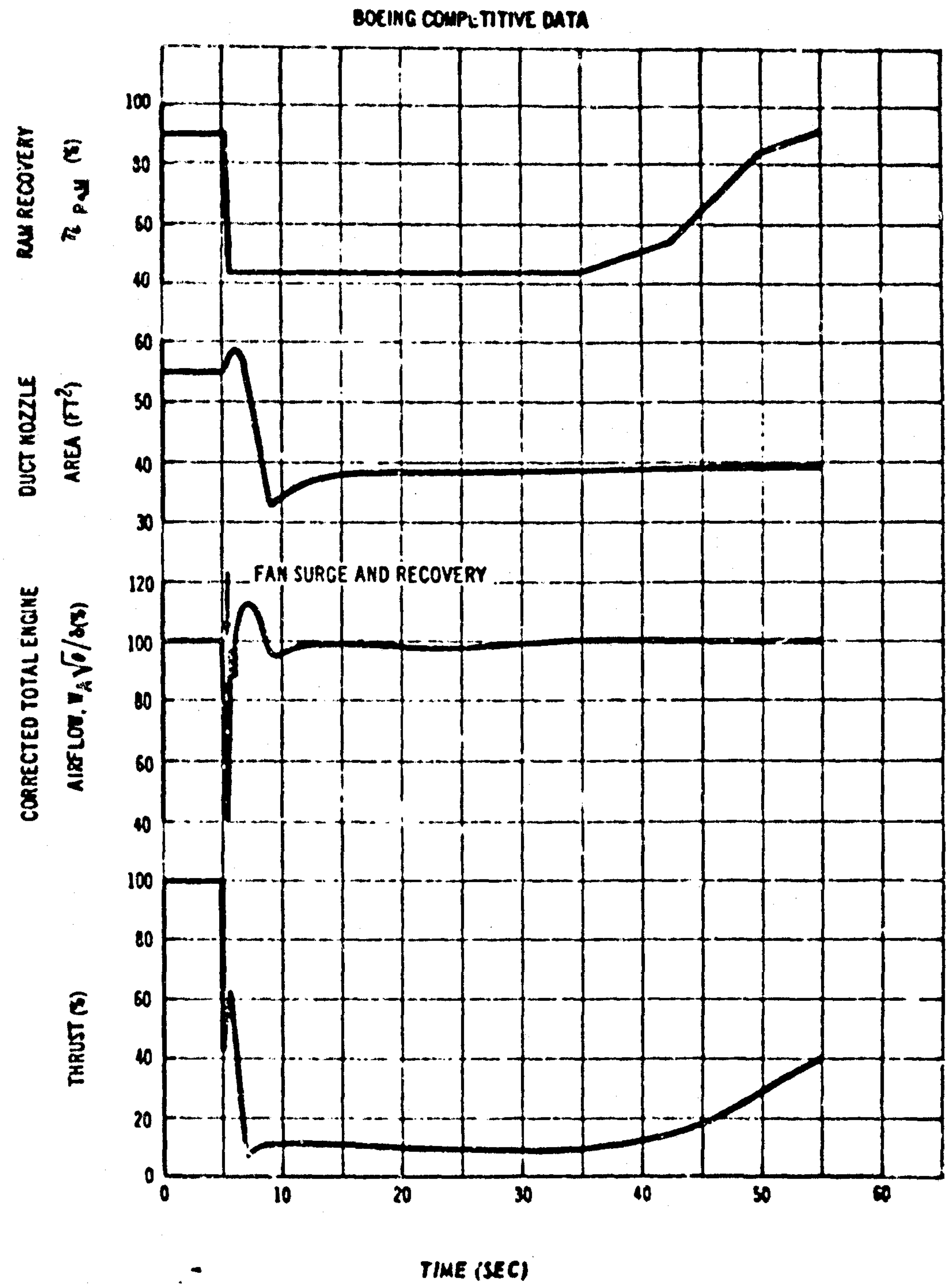

Figure A-34. JTF17 Inlet Unstart and Restart at Cruise

ILLUSTRATIONS

Tables

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