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SMALL ENGINE COMPONENT TECHNOLOGY (SECT) STUDY FINAL REPORT

By B. Singh
TELEDYNE CAE
Turbine Engines

Date for general release March 31, 1991

Prepared For:
National Aeronautics And Space Administration
Lewis Research Center
And
U.S. Army Aviation Research and
Technology Activity Propulsion Directorate

Contract NAS 3-24541

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16. Abstract

A study has been conducted for advanced small (450 - 850 pounds thrust, 2002 - 3781 N) gas turbine engines for a subsonic strategic cruise missile application, using projected year 2000 technology. An aircraft, mission characteristics and baseline (state-of-the-art) engine were defined to evaluate technology benefits. Engine performance and configuration analyses were performed for two and three spool turbofan and propfan engine concepts. Mission and Life Cycle Cost (LCC) analyses were performed in which the candidate engines were compared to the baseline engine over a prescribed mission. The advanced technology engines reduced system LCC up to 41% relative to the baseline engine. Critical aerodynamic, materials and mechanical systems turbine engine technologies were identified and program plans were defined for each identified critical technology.

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1.0 SUMMARY

The objectives of this study were to identify high payoff small turbine engine technologies for year 2000 applications and provide technology plans for guiding future research and technology efforts. The study was based on a subsonic strategic cruise missile capable of flying at Mach 0.7 to 0.9 from sea level to 12.2 KM(40,000 ft.) with a 7408 KM (4000 NM) range. A current state-of-the-art engine (2758N, 620 lbs thrust at sea level, Mach 0.7) was selected as the baseline engine.

An engine cycle performance analysis was conducted at sea level, Mach 0.7 using the projected year 2000 technology. Four engine concepts were evaluated with a variety of component configurations: two spool turbofan, three spool turbofan, two spool propfan and three spool propfan. Four candidate engine cycles, representing each engine concept, were selected for system analysis. These engines ranged from 1371 to 1649°C (2500 to 3000°F) turbine rotor inlet temperature and 26:1 to 45:1 overall pressure ratio. The engines were selected primarily on the basis of low SFC and component configuration considerations. The lowest SFC was achieved with the three spool propfan engine cycle: approximately 0.051 Kg/HR/N (0.5 lb/hr/lb), a 50% reduction relative to the base-line engine. The three spool turbofan engine provided approximately 35% reduction in SFC.

Each candidate engine was subjected to mission analysis and compared to the baseline engine mission performance. Engines and air vehicles were scaled to satisfy the 7408 KM (4000 NM) mission range. All advanced candidate engines satisfied the required 7408 KM (4000 NM) range. The baseline engine, scaled to 4448 Newton (1000 lb) thrust, failed to meet this requirement by 30%. Missile weight ranged from 782 Kg (1725 lbs) for the three spool propfan to 1483 Kg (3271 lb) for the baseline engine. The reduction in missile weight and size increased the launch aircraft missile carrying capability up to 47% based on missile weight and 26% based on the missile diameter relative to the baseline engine.

A life cycle cost (LCC) analysis was performed, to compare the advanced technology engines to the baseline engine. The results indicate that the advanced technology engines provide up to 41% reduction in system LCC. A turbine cooling penalty analysis indicated 15% increase in LCC relative to the uncooled turbine. Recuperated configurations were evaluated, and found to be noncompetitive.

High payoff engine technologies were identified and ranked. The technologies identified as critical to the success of the year 2000 systems are: ceramic composite radial inflow turbine, light weight missilized propfan gearbox and high speed bearings/seals. Technology plans were prepared for these and other identified important technologies.

2.0 INTRODUCTION

Many research and development programs have resulted in an extensive data base for medium to large gas turbine engines. However, small engines have not attracted the same interest and funding. This has created a wide technology and performance gap between small (.23-1.59 Kg/sec, 0.5-3.5 lbs/sec airflow) and large engines; small engine components exhibit lower efficiency and temperature levels. Technology transfer is limited, because aerodynamic and structural design and manufacturing techniques of large engines can not directly be scaled to small engines. However, technology programs directed to small engines can provide up to 50% fuel consumption improvements for the year 2000 technology engines.

The primary objectives of the Small Engine Component Technology (SECT) study were to identify high payoff technologies and provide technology plans for year 2000 technology turbine engines, defined as covering the 890-4448 Newton (200 to 1000 lbs) thrust range. This study was funded by the NASA Lewis Research Center and US Army Aviation Research and Technology Activity - Propulsion Directorate. Technology payoffs are based on system life cycle cost evaluation. Technology plans include detailed schedules.

The SECT study was directed to an advanced strategic subsonic cruise missile application. The size, configuration and general performance needs of rotorcraft, tactical cruise missile and APU engines are congruent with those of the strategic cruise missile engine. Because many of the required technologies are common, emphasis was placed on technology transfer to these applications.

The SECT study consisted of the following tasks:

(i) Task I - Selection of evaluation procedures and assumptions. This effort focused on defining the study methodology and major assumptions.

(ii) Task II - Engine configuration and cycle analysis. This task encompassed component technology projections, engine cycle analysis, component definition and selection of candidate engines.

(iii) Task III - System performance evaluation. Mission and life cycle cost analyses were conducted for the candidate engines.

(iv) Task IV - Technology plan. High payoff technologies were identified and ranked during this task. Technology plans were generated for the high payoff technologies.

This report presents the results of this study.

3.0 SELECTION OF EVALUATION PROCEDURES AND ASSUMPTIONS

The Small Engine Component Technology (SECT) study assumptions and groundrules included definition of the baseline application, and definition of: mission requirements, reference air vehicle and engine, basic assumptions and methodology. The primary mission is a strategic subsonic cruise missile, with the required engine technology to be available by the year 2000.

3.1 Methodology Definition

A methodology was established to conduct the engine configuration and cycle evaluation, system performance evaluation and to establish a SECT plan; the logic flow is shown on Figure 1. The methodology integrates mission requirements, projected components and materials technology, engine cycle/configuration analysis and aircraft definition to identify the high payoff technologies. The main elements of the methodology are:

(i) Assumptions and groundrules

(ii) Mission requirements

(iii) Reference vehicle

(iv) Reference state-of-the-art engine

(v) Engine cycle analysis

(vi) Engine concepts and configuration

(vii) Mission analysis

(viii) Life cycle cost analysis

(ix) Technology identification and prioritization

(x) Technology transfer

(xi) Technology plans

3.2 Assumptions And Groundrules

Assumptions and groundrules were established to provide consistent advanced technology results. These included engine parameters, engine operating conditions, engine installation and LCC parameters. These assumptions were based on advanced engine cycles, concepts and materials to achieve large SFC reductions. The main assumptions are summarized in the following paragraphs. A list of symbols and abbreviations is contained in Appendix A.

Range of Thermodynamic Parameters

The ranges of thermodynamic parameters selected were:

Bypass Ratio = 2.0 to 6.0 (Turbofans only)
Turbine Inlet Temperature = 1149-1927°C (2100 to 3500°F)
Overall Pressure Ratio = 8.0 to 45.0
Propfans: Single and Counterrotation

This technology span was based on the mission requirements, year 2000 technology projections and Teledyne CAE's past experience:

(i) Subsonic flight speeds dictate that bypass ratios be between 2.0 and 6.0 for low SFC, depending upon the turbine inlet temperature and installation restrictions.

(ii) Turbine inlet temperature impacts the core engine size, hence allowable bypass ratio, therefore, year 2000 engines demand high temperatures. The low turbine inlet temperature limit was selected to reflect current or near term operational technology levels.

(iii) High overall pressure ratio (OPR) is desireable for SFC considerations. However, excessive OPR's can result in reduced component size, increased engine complexity and cost. An OPR limit of 45:1 provides achieveable goals in the year 2000 technology engines.

(iv) Single and counterrotating propfans provide a means of significant SFC improvements for the subsonic strategic cruise missile mission.

Engine Operating Environment

The engine operational environment conditions are:

Altitude = 0-12.2 KM (0-40,000 ft.)
Mach No. = 0.5 - 0.9
Distortion = KD2 $\geq$ 1800 (See Appendix A for definition)

The subsonic strategic cruise missile is air launched, performs a first cruise segment at 10.7 KM (35,000 ft.) and descends to sea level for the final cruise. The missile was assumed to have a flush type inlet (due to reduced RCS requirements), demanding an engine distortion tolerance capability.

Engine Installation Losses

The assumed engine installation losses are:

Inlet Recovery: 0.96
Bleed Air Extraction: 0.0
Power Extraction: 7.5 KW (10.0 HP)

Fuel:

Current cruise missiles use JP-10 as the primary fuel. Thus, JP-10 was selected to provide relative technology payoffs, and to allow relation of cruise missile technology to the technology transfer SECT applications, which most probably will not use slurry fuels.

Slurry fuels were considered in defining engine configurations, but were not considered for detail analysis as they are outside the SECT work scope. Some of the uncertainties and problems with various slurry fuels are: ill defined combustion chemistry, liquid combustion product deposit on the engine components, engine component abrasion due to combustion products, fuel management system (tank, pumping, valving) complexity and interaction of combustion products with composite materials.

Technology Drivers

Significant strategic cruise missile system performance (range) improvements will require, in decreasing priority order: SFC reduction, and increases of thrust/frontal area, specific thrust and thrust/weight relative to current engines.

The assumed technology drivers (goals) were:

SFC: .051-.082 Kg/hr/N (0.5 - 0.8 lb/hr/lb)
Thrust/frontal area: .035-.07 N/MM^2 (5-10 lb/in^2)
Cost: $22.5-45/N ($100-200/lb) thrust
Thrust/airflow: 294-981 N/Kg/sec (30-100 lb/lb/sec)
Thrust/weight: 44.1-98.1 N/Kg (4.5 - 10 lbs/lb)

Trade Factors and Preliminary Selection Criteria

The following trade factors were defined to be used as preliminary engine selection criteria:

TRADE FACTORS	VALUES (%/%)
$\partial W/\partial$ SFC	1.5 - 3.0
$\partial W/\partial$ (FN/WE)	0.2 - 0.5
$\partial W/\partial\ \rho E$	0.1 - 0.25

Where, W = vehicle launch weight
SFC = specific fuel consumption
FN = engine max thrust
WE = engine weight
ρE = engine density

These representative trade factors were derived from past studies for a typical subsonic strategic cruise missile mission. Since the SFC trade factor is overwhelmingly dominent over the other two, the preliminary engine selection process focused entirely on SFC.

Life Cycle Cost (LCC) Assumptions & Procedures

The SECT system LCC analysis was conducted based on the following assumptions:

(i) Constant 1985 dollars
(ii) 6000 production units
(iii) 600 units per year production rate
(iv) 5 year scheduled maintenance interval
(v) 20 year engine life. An engine life starts at the time of delivery and expires at the end of 20 years, with scheduled maintenance every five years.
(vi) Fuel cost: $10 per gallon for JP-10. $35 per gallon was also used for sensitivity analysis.
(vii) Cost categories: Engine development
Engine Acquisition
Engine maintenance
Fuel
Vehicle acquisition

Development and production unit costs were computed through the use of the RCA PRICE H model. Engine maintenance costs were computed through the use of the APSICOST MOSC (Missile Operational and Support Cost) model, developed under USAF/AFWAL sponsorship. The computer model's conceptual flow chart is presented in Figure 2.

3.3 Mission Requirements

A representative subsonic strategic cruise missile mission (Figure 3) was selected to evaluate each advanced technology candidate engine and the baseline engine. The primary characteristics of the mission are:

(i) Altitude Launch: air launch at 11.28 KM (37,000 ft.), Mach 0.7; accelerate to cruise Mach 0.9 at 10.67 KM (35,000 ft.)

(ii) Altitude Cruise: cruise at 10.67 KM (35,000 ft.), Mach 0.9 for 4630 KM (2500 NM range).

(iii) Descent: descend to sea level, Mach 0.7 without range gain or fuel used.

(iv) Sea Level Cruise: cruise 2778 KM (1500 NM) at Mach 0.7.

(v) Termination: Mission terminates after achieving 7408 KM (4000 NM) total range.

(vi) Payload: 181.4 Kg (400 lbs).

This mission was chosen based on the following considerations:

(i) Previous studies indicated 7408 KM (4000 NM) range puts launch aircraft beyond most enemy threat boundaries, decreasing the attrition of this expensive weapon system (not included in life cycle cost impact of SECT technology). Mach number is consistent with the range requirements.

(ii) The mission range represents a significant improvement over current systems.

(iii) Longer ranges than 7408 KM (4000 NM) will require a larger vehicle, thus resulting in an increased engine size (larger than 4448 Newton, 1000 lbs - outside the scope of this program).

(iv) A very large missile will be difficult to air launch.

3.4 Aircraft Definition

The selected air vehicle (Figure 4) and its characteristics were based on a NASA Langley study (Reference 1). The vehicle features a relatively high drag-rise Mach number of about 0.95. The basic vehicle characteristics are:

Aspect Ratio: 2.446
Wing Leading Edge Sweep: 58°
Wing Section: NACA 65A006
Vehicle Length/Diameter: 8.65

The baseline vehicle parameters are:

Vehicle Weight	=	2495 Kg (5500 lbs.)
Wing Area	=	4.93 M^2 (53.1 ft^2)
Wing Span	=	3.47 M (11.4 ft.)
Fuselage Length	=	6.77 M (22.2 ft.)
Fuselage Diameter	=	.78 M (2.56 ft.)

The vehicle planform and side views are depicted in Figure 4. The fuselage features a circular cross-section; Wings are attached at the center line of the fuselage. The vehicle was scaled for each advanced technology candidate engine to satisfy the mission range requirement of 7408 KM (4000 NM).

The vehicle launch and fuel weights were estimated based on the engine and air inlet size and weight. The vehicle was required to satisfy the following two requirements:

TOTAL FUSEAGE VOLUME = (PAYLOAD+STRUCTURAL+ENGINE+INLET + FUEL) VOLUME

LAUNCH WEIGHT = (PAYLOAD+STRUCTURAL+ENGINE+FUEL) WEIGHT

The vehicle drag characteristics are representative of an advanced cruise missile vehicle, Figure 5.

The engine is mounted in the aft section of the fuselage. Engine air is supplied through a bottom mounted flush inlet. The propfan engines use pusher counter-rotating props mounted aft of the engine and located behind the vertical tail.

3.5 Baseline Engine Definition

The reference engine is based on the current state-of-the-art technology. The baseline engine is a two spool turbofan featuring a bypass ratio of 1.1 (Figure 6), scaled from 2758 to 4448N (620 to 1000 lb) thrust to achieve maximum range. The LP spool consists of a two-stage fan driven by a two-stage uncooled turbine. The HP spool features a four stage (3 axial plus one centrigugal) compressor driven by high load capacity single stage uncooled turbine. An annular slinger combustor provides a compact engine installation. The exhaust system mixes by-pass and core flows, discharging through a convergent nozzle. Basic engine performance is summarized in Table 1 and engine materials are listed in Table 2.

4.0 ENGINE CONFIGURATION AND CYCLE ANALYSIS

Engine cycle values were studied parametrically to identify and define candidate cycles for the advanced subsonic strategic cruise missile system. The study was conducted using the assumptions and ground rules of Section 3.0 nd preliminary component projections at a sea level Mach 0.7 flight condition. This flight condition was identified as the most critical from fuel usage considerations, as more than 50% of the missile fuel is used at sea level cruise.

Detailed component aerodynamic, structural and materials technology projections were later made and integrated into the candidate engines after the thermodynamic parameter ranges were reduced to a manageable number and high payoff combinations were identified. They verified the accuracy of the initial, simpler projections.

4.1 Preliminary Technology Projections

Preliminary component efficiency projections were made to conduct broad engine cycle parametric analysis. These efficiency projections were defined in a format to facilitate and simplify engine cycle analysis, and later verified for the final selected engine cycles. The primary objective was to generate realistic and consistent engine parametric cycle data and provide relative cycle performance assessment without detailed component design and analysis.

Axial and centrifugal compressor polytropic efficiency projections are presented in Figure 7 as a function of compressor exit corrected airflow. These projections were derived from historical data and projected to the year 2000. Higher corrected airflow data are shown in Figure 8, and show that the axial and centrifugal efficiency curves cross over at approximately 2.5 Kg/sec (5.5 lbs/sec) corrected airflow. The axial curve passes through the Energy Efficient Engine (E^3) compressor efficiency level at 4.5 Kg/sec (10.0 lbs/sec) corrected airflow (Reference 2).

The uncooled turbine efficiency projections were derived in a similar manner. Adiabatic efficiency is presented in Figures 9 and 10 as a function of turbine inlet corrected airflow for radial and axial turbines respectively. Axial turbines reached E^3 engine efficiency levels at approximately 5.0 Kg/sec (11.0 lbs/sec) corrected airflow (Reference 3).

4.2 Engine Cycle Analysis

The parametric analysis covered a broad range of cycle parameters to ensure that best engine cycles are selected for the Task III system analysis. The strategic cruise missile engine size is relatively small, 0.23 to 1.81 Kg/sec (0.5 to 4.0 lb/sec) core corrected airflow. This indicates a desireability of uncooled turbines. However, a cooled turbine analysis was also conducted to provide trade data (Section 4.8). The cycles were sized for a baseline thrust at approximately 3336 Newton (750 lbs) at sea level, Mach 0.7 with 0.96 inlet pressure recovery. Four basic engine concepts were evaluated during this study:

4.2.1 Two-Spool Turbofan

It was assumed that the maximum overall pressure ratio (OPR) achievable in a two-spool turbofan is 30:1. The parametric data are presented in Figure 11. Fan pressure ratio was optimized for each combination of bypass ratio, turbine rotor inlet temperature (TRIT) and overall pressure ratio. The data indicate the optimum (lowest SFC) thermodynamic parameter ranges to be:

TRIT	1149-1649°C (2100-2500°F)
OPR	25-30
Bypass Ratio	4-6

4.2.2 Three-Spool Turbofan

Engine thermodynamic cycle data are presented in Figure 12 for a three-spool turbofan engine. The three-spool turbofan provides the capability of achieving higher OPR and lower SFC than a two-spool turbofan. Fan pressure ratio was optimized for each combination of bypass ratio, TRIT and OPR. SFC improvements of up to 7% were realized relative to the two-spool engines. Data did not indicate a significant SFC improvement above OPR of approximately 36:1. The optimum (lowest SFC) thermodynamic parameters are:

TRIT	1371-1649°C (2500-3000°F)
OPR	36-45
Bypass Ratio	4-6

4.2.3 2-Spool Propfan

Recent developments in propfan technology have proven that they can achieve high efficiency levels at high subsonic Mach numbers (0.8-0.9). Therefore, the NASA propfan concept was evaluated for a strategic cruise missile application in order to achieve significant gains in range. Single-rotation propeller (SRP) and counter-rotation propeller (CRP) concepts were evaluated for the prescribed mission. CRP permits higher propeller loadings (HP/D^2) and achieves higher efficiency levels than the SRP. Three CRP configurations (4X4, 5X5 and 6X6 bladed) were evaluated for the SECT study. CRP performance was generated using the Hamilton Standard performance decks. Propfan efficiency as a function of power loading and tip speed is presented in Figure 13 for the selected (4X4) CRP configuration at sea level Mach 0.7. The selected propeller characteristics are:

No. of Blades	4X4
*Power Loading (HP/D^2)	73.4-342.5 Kw/M^2 (30-140 HP/Ft^2)
Tip Speed	243.8 M/sec. (800 Ft./Sec.)
Efficiency	0.88

*Depending upon the cruise horsepower requirements.

A fixed pitch counter-rotation propeller configuration was used for simplicity; the variable pitch mechanism is very costly and complex, therefore, undesirable for the unmanned applications. Preliminary fixed pitch CRP evaluation did not indicate any significant reduction in propeller efficiency down to 80% power; minimum cruise power required was at or above 80% (Section 5.1).

These characteristics were used to generate engine parametric data for two-spool propfan engine concepts. SFC and specific thrust are presented as a function of OPR and TRIT in Figure 14. These data indicated very significant (up to 29%) improvements in SFC relative to the two-spool turbofan, and up to 50% relative to the baseline engine. The optimum (lowest SFC) thermodynamic parameters are:

TRIT	1371-1649°C (2500-3200°F)
OPR	22-30

Specific thrust varies very significantly with TRIT, e.g., a 45% increase in specific thrust is observed at 1649°C (3000°F) TRIT relative to 1371°C (2500°F).

4.2.4 Three-Spool Propfan

Propfan engine parametric data were generated using the CRP performance described in Section 4.2.3. Parametric data are presented in Figure 15 as a function of TRIT (1371-1649°C) and OPR (30-45). These data indicate further reduction in SFC relative to the lowest SFC achievable from the two-spool propfan cycle (Figure 14). However, specific thrust is reduced up to 15% relative to the 2-spool propfan cycles due to the increased OPR.

4.3 Engine Cycles For Component Analysis

Three engine cycles were selected for the component design effort. These cycles based on Section 4.2 data analysis are: two spool turbofan with TRIT = 1371°C (2500°F) and overall pressure ratio (OPR) = 25; three spool turbofan with TRIT = 1649°C (3000°F) and OPR = 45; and two spool propfan with TRIT = 1649°C (3000°F) and OPR = 22. Specific design requirements for these three cycles are shown in Table 3. The three spool propfan cycle was not selected for component design analysis because the three engine concepts encompass the component requirements for this engine.

The conceptual component design and analysis was carried out for the specific design requirements (pressure ratio, corrected flow, and work) established by these cycles. This approach provided realistic component configuration comparisons for the same design requirements.

4.4 Component Aerodynamics

The component configurations shown in Table 4 were designed for each engine concept based on the flow and work defined by the optimized cycles. The components were mixed and matched as required. The projected year 2000 component aerodynamic technologies used for these designs are discussed in this section.

4.4.1 Fan/Compressor Technology

The compressor technology assessment was based on a realistic compressor data base and a generic computer code. Compressor efficiency is a function of parameters such as airflow, pressure ratio, tip speed, inlet radius ratio, specific speed, backward curvature, and diffuser radius ratio, thus is difficult to present in compact form. Performance characteristics were projected for the year 2000 for axial and centrifugal compressor stages.

4.4.1.1 Technology Projections

Teledyne CAE projects an increase of 5-7 points in compressor adiabatic efficiency by year 2000. Figure 16 shows projected performance as a function of temperature coefficient for axials, and as a function of stage pressure ratio for centrifugal stages. These axial efficiency levels are adjusted for hub/tip ratio, Figure 17. Specific speed corrections are also applied to centrifugal stage efficiency levels, Figure 18. Figure 19 addresses the engine size penalty on the compressor performance. Multi-stage compressor performance was predicted using these figures for each component stage. The compressor performance projections shown in these figures are consistent with the advanced structural and materials guidelines projected for year 2000 applications (sections 4.5 and 4.6).

Compressor performance improvements can be achieved by enhancing both aerodynamic analytical modeling and structures/manufacturing. Aero modeling will be improved by the use of 3-D aero codes like the Teledyne CAE/Denton code now being integrated into the compressor design system. The use of advanced codes such as this for flow analysis and blading design will improve efficiencies, allow higher stage loadings and minimize the penalties associated with small blade passages.

Structural modeling needs improved capability for more accurate blade untwist and uncamber prediction, for higher tip speeds, and for tighter tip clearances. Improvements in these areas will produce reduced size corrections and higher pressure ratios per stage, thereby increasing efficiencies.

In the area of materials, it will be necessary to handle smaller shaft diameters and higher tip speeds so that hub/tip radius ratio corrections are improved, better stage pressure ratios are obtained and efficiency increased.

Improved manufacturing will result in airfoils with smoother surface finish, smaller leading edge and fillet radii, and closer blade tolerances. These items will also improve efficiencies and reduce the penalties imposed by small size.

4.4.1.2 Configurations

Specific fan/compressor configuration designs were evaluated for two and three-spool turbofans and two-spool propfan cycles. These configurations provided enough information to satisfy the requirements for the three-spool propfans, therefore, it was not considered necessary to separately evaluate the compressor designs for this cycle. Temperature coefficient and blade height limitations were used to provide practical design concepts, Table 5. The components were designed at sea level, Mach 0.7 flight condition.

4.4.2 Turbine Technology

Advancements in the development of high temperature turbines are required to achieve the performance and structural durability goals for the year 2000 technology small turbine engines. Because of the performance, cost and complexity of small cooled turbines it was decided to baseline uncooled turbines for the strategic cruise missile application. Axial, mixed flow and radial turbine configurations were evaluated and efficiency levels projected to the year 2000.

4.4.2.1 Technology Projections

Teledyne CAE projects an increase of 5 to 6 points in limited life uncooled axial turbine efficiency at temperature levels of 1371°C (2500°F) to 1927°C (3500°F). Similar improvements are projected for the radial turbines. Generic projected turbine efficiency levels are presented in Figures 20 through 22 for axial, radial and mixed flow configurations. The projected performance is presented for three turbine inlet corrected flow levels as a function of loading ($\Delta H/\theta cr$), U/Co and speed (RPM). Speed has been used as an independent parameter. However, an optimized turbine speed may not be feasible for a specific design due to its impact on the compressor design. Therefore, speed selection was based on the compressor-turbine performance considerations.

These projections are based on improved computational analytical methods (such as three-dimensional airfoil design techniques and three-dimensional viscous flow analysis), high Mach number blades, vane-blade interaction analysis, active/passive tip clearance control, high tip speeds (material improvements) and manufacturing technology improvements to provide improved surface finishes, Table 6. In addition, the losses due to secondary flows and low-aspect-ratio blade rows (necessitated by high loadings, low solidities and AN^2 limits) will be reduced through the use of fully three-dimensional aerodynamic design tools. The AN^2 limit is projected to be 2.15×10^4 M^2-Rev/sec^2 (1200×10^8 in^2-RPM^2) for ceramic composite axial turbines; typical current values are 0.72-1.43×10^4 M^2-Rev^2/Sec^2 (400-800×10^8 in^2-RPM^2). Additional technology projection groundrules and limitations are summarized in Table 7.

4.4.2.2 Configurations

Specific turbine configuration designs were evaluated for two and three-spool turbofans and two-spool propfan cycles (Section 4.3). These configurations also provide enough information to satisfy the requirements for the three-spool propfans.

4.4.3 Combustor Technology

The candidate SECT engines use overall pressure ratios up to 45:1 and high turbine inlet temperatures to optimize overall engine performance. At high temperatures, the durability and life of the turbine components are strongly dependent on combustor exit temperature gradients. The radial temperature profile must meet the structural requirements of the rotating component, while the circumferential temperature gradient (or pattern factor) must be minimized to reduce thermal stresses and cooling requirements of the static structure. Consequently, the combustors will be required to have excellent exit temperature quality, and to retain this quality throughout the life of the engine. Control of combustor exit temperature gradient will require accurate prediction and control of the aerothermodynamics and stoichiometry of the primary combustion zone through the use of improved design tools. In addition, light weight and high temperature capability materials will be required. Combustor performance characteristics were projected to meet the year 2000 system goals.

4.4.3.1 Technology Projections

Small turbine engine combustors currently operate with pattern factors in the 0.2 to 0.3 range at temperature-rise-to-inlet-temperature ratios ($\Delta T/T4$) in the order of 1.4. With the high cycle pressures and temperatures required of the SECT engines, turbine durability and life goals will require pattern factors below 0.15. Combustor performance projections for the year 2000 are compared in Table 8 with the current values.

Cooled Configurations

The year 2000 engines will require increased temperature rise across the combustor (Figure 23) thus adding to the combustor cooling complexity. The cooling effectiveness achieved on current film cooled combustors (0.4 to 0.6) is expected to reach 0.9 by the year 2000 via advanced augmented surface convection film cooling concepts and advanced materials.

Durability, reliability and/or life goals of advanced engines conflict with performance goals such as increased thrust-to-weight ratio and low SFC. This is particularly true in the combustor, where the high cycle pressures and temperatures needed to attain overall engine performance goals necessitate the use of advanced cooling, materials and fabrication techniques - which add complexity and (in some cases) weight. While advanced cooling concepts do increase cooling effectiveness, (Figure 24), the ability of current heat transfer analysis to accurately predict these improvements is hampered by the limited detailed knowledge of the local boundary conditions. Advanced cooling concepts, augmented by improved analytical models integrating the combustor flowfield analysis and the liner wall heat transfer mechanism, will allow more effective use of cooling air and provide a more detailed, accurate prediction of liner temperatures. These improved combustor flowfield models are expected to alleviate this problem. The added complexity and weight are illustrated in Figure 25, where the weight of the advanced convention-film cooled configuration is three times that of the sheet metal louvered film concept. Part of this weight increase is offset by the trend toward smaller combustors (Figure 26).

There has been a general trend toward a lower pattern factor (PF) over the years (Figure 27) and this trend is expected to continue. A push towards reduced pressure loss (Figure 28), with an attendant degradation in mixing, flowfield and gradient control will be balanced against exit temperature gradient requirements.

Uncooled Materials

Significant advances are required to develop high temperature capability materials for combustors. These include fiber reinforced superalloys, ceramic composites and carbon-carbon. The ceramics and carbon-carbon are very attractive since they offer very light weight structures with material temperature capabilities in the 1760 to 2404°C (3200 to 4000°F) range, Table 11.

These advanced materials will eliminate the need of combustor liner cooling requirements thus resulting in higher and more uniform wall temperatures. These (uncooled) hot walls will provide two major benefits to the combustion process:

(i) The wall quenching associated with cooled walls and/or cooling films will be eliminated, or at least significantly reduced. In small combustors where surface area to volume ratio is high, wall quenching can represent a significant loss in combustor performance.

(ii) Heat loss from the flames (due to radiation) to the hot walls will be reduced. This heat loss reduction will result in increased reaction rate and thus improved combustor performance.

4.4.3.2 Selected Configurations And Flowpaths

Specific combustor designs were evaluated for two spool turbofan, three spool turbofan and two spool propfan engine configurations. In order to provide consistent data, the following design goals/guidelines were assumed:

- o Combustor Pressure Loss $\leq$ 2%
- o Pattern Factor $\leq$ 0.15
- o Fuel (Heating Value) JP-10: 42,100 KJ/Kg (18,100 Btu/Lb)
- o High efficiency and wide range stability.

Three basic combustor configurations were considered; annular slinger with centrifugal fuel injection, straight through or axial annular with atomizing fuel nozzles and reverse flow annular, with atomizing nozzles, Figure 29. Combustor geometry and size are dependent on the high pressure compressor/turbine configurations. Therefore, specific conceptual combustor design configurations were selected to provide compact engine flowpaths for the selected high pressure compressor/turbine combinations.

Based on this configuration analysis, the slinger type combustor design presented in Figure 29 was selected to provide compact engine configurations for two spool turbofan, three spool turbofan, and two spool propfan. Combustor loadings are light to moderate: residence time 5.2 to 14.8 milliseconds and heat release rate 35.3 to 97.8 X 10^{-5} J/Sec M^3 Pa (4.3 - 11.9 MBTU/hour/cubic foot/atmosphere). The resulting combustor loadings and slinger design concepts will provide wide range for high performance and adaptability to the slurry fuels.

4.5 Structural Technology

Aggressive performance goals for engines in the year 2000 will require significant increases in tip speeds and significant decreases in component weights, obtained through the use of increased strength-to-weight materials and increased design efficiencies. Design efficiency will be increased by tailoring the design to the selected material capabilities, using advanced structural analysis methods. Increased use of inelastic stress analyses and composite material stress and life analysis computer codes are projected.

Engine static component durability requirements generally have no significant interactions with the engine performance requirements, but many static components contribute significantly to engine weight. The design of each static structure component must be tailored to take full advantage of the properties of the selected material. This structural tailoring is particularly important for composite structures. Cost is a governing concern in missile engines, and difficult to estimate for these unproven materials.

This section discusses the structural aspects, approaches, assumptions, design limits and the design requirements for the year 2000 technology engines. The structural guidelines focus primarily on the rotating components as they provide the maximum technology payoffs. In addition, technologies for mechanical systems such as bearings, seals, shafts and the propfan gearbox are also addressed.

4.5.1 Fan/Compressor Structural Technology

Selection of rotating component mechanical design concepts and materials technology is critical to the study, since durability considerations for these components will limit performance parameters such as tip speed, aspect ratio and hub/tip ratio. Preliminary structural design studies were conducted for axial, mixed flow and centrifugal compressor rotating components to identify the projected limits for these parameters. These studies identified the critical structural design requirements indicated in Table 9.

Axial/Mixed Flow Fan/Compessor

Figure 30 shows projected maximum allowable tip speed as a function of hub/tip ratio (based on average diameters) and aspect ratio for axial and mixed flow fan/compressor rotors. The baseline engine rotor characteristics are also superimposed on this figure. The limits are based on burst margin and/or creep/rupture requirements. Obtainable tip speed increases for year 2000 technology are in the range of 40 to 80 percent. These technology improvements are based on advanced materials, principally materials with a density of 5.26 X 10^3 Kg/M^3 (0.19 lbs/in^3). Specific material selection will be based on the fan/compressor maximum operating temperature. For example, reinforced aluminum can be used at temperatures to 315°C (600°F) and composites such as carbon/polymide to approximately 427°C (800°F). Reinforced titanium is projected to have capability to 704°C (1300°F). Additional limitations imposed are:

- (i) 965-1138 X 10^6 Pa (140-165 KSI) yield strength for the advanced materials
- (ii) No vibratory restrictions
- (iii) An assumed blade thickness/chord ratio of 9% at root, linearly tapering to 2.5% at the tip with a constant chord blade.

Component design is limited by the composite matrix strength, namely the inter-laminate and in-plane shear stresses in the fibers. A second critical limit is the transition region from radial stress to circumferential stress in the fibers.

Centrifugal/Mixed Flow Compressor

The centrifugal compressor configuration is often limited by creep/rupture concerns. Since the disk bore temperature parallels that of the compressor exit, the bore is usually the limiting location for this component. These concerns are aggravated by the bore diameter requirements imposed if a low pressure spool shaft must pass through these components.

Figure 31 presents projected allowable tip speeds for several materials as a function of disk bore temperature. The utilization of advanced titanium aluminide alloys, nickle based alloys and carbon-carbon are projected to provide 792-914 M/Sec (2600-3000 ft/sec) tip speed capability. Allowable tip speeds are indicated to be as high as 25% greater than the current state-of-the-art.

These projected improvements in tip speed are based upon a 20% improvement in ultimate strength capability of metallic materials through the use of advanced nickel alloys as well as titanium aluminide alloys. These gains can be accomplished through the rapid solidification process which is currently under development.

For advanced composite materials, a 2X improvement in shear strength capability of carbon/carbon or similar material is required: interlaminate shear stress capability of 6.89×10^6 Pa (1 KSI) versus 3.45×10^6 Pa (0.5 KSI) and 103×10^6 Pa (15 KSI) for the in-plane shear stress capability are suitable goals. These improvements are required in conjunction with a low temperature oxidation/corrosion coating.

The centrifugal/mixed flow component designs are typically stress limited by the disk bore utilizing a disk burst margin of 17%.

4.5.2 Turbine Structural Technology

Advanced turbine components are projected to require significant improvement in structural capabilities over current turbine technology. Turbine disks will have to increase in rim speed and turbine blades to increase in average metal temperatures. Improvements in materials and continued development of structural design methodology will ensure the achievement of these advanced capabilities, while maintaining the required reliability to meet the projected mission usage goals.

The fundamental structural design requirements which must be addressed during any turbine design and development are outlined in Table 10. The specific design requirements of burst, creep/rupture, some low cycle fatigue, high cycle fatigue, damage tolerance, maneuver loads, and containment are the key considerations that establish a turbine component's structural integrity. Of these, creep/rupture and low cycle fatigue (LCF) are the primary drivers. For disks and blades, the critical structural parameters that in turn drive creep/rupture and low cycle fatigue, are rim speed, AN^2 (annulus area x speed squared) and average metal temperature.

Axial/Mixed Flow Turbines

The mixed flow turbine technology is represented by additional design parameters such as the inlet and outlet lean angles. A technology capability for AN^2 of $2.15\text{-}3.5 \times 10^4$ $M^2\text{-}REV^2/SEC^2$ ($1200\text{-}1950 \times 10^8$ in^2 RPM^2), using advanced ceramic composites and carbon-carbon materials, are expected. The corresponding tip speeds vary from 792 to 914 M/Sec (2600 to 3000 ft/sec). Current technology limitations for tip speed and AN^2 are 617 M/Sec (2200 ft/sec) and 1.43×10^4 $M^2\text{-}Rev^2/Sec^2$ (800×10^8 in^2 RPM^2) respectively.

Advanced materials such as ceramic composites and silicon carbide/silicon carbide are key technologies that are required in achieving these turbine technology goals. A 5 to 10% improvement in basic tensile strength capability with an operating temperature 1649-1927°C (3000-3500°F) was assumed for the ceramic composite materials. A 200% improvement in carbon/carbon shear stress capability is required to attain these projections. The improvement in shear stress capability reflects 6.89×10^6 Pa (1 KSI) interlaminate shear stress and 103×10^6 Pa (15 KSI) in-plane shear stress capability. Along with this, a high temperature oxidation coating capable of withstanding 1649-1927°C (3000-3500°F) is required.

The turbine blades are stress limited by the parameter AN^2. This is basically set by the airfoil twist and by the airfoil lean for the mixed flow rotors. The disk is also stress limited by tip speed parameter.

Radial Turbines

The tip speed versus turbine rotor inlet temperature relationship is used to establish the turbine blade's creep/rupture limitation. Advanced technology carbon-carbon tip speed projections range from 792 to 914 M/Sec (2600 to 3000 ft/sec) up to 1927°C (3500°F) turbine rotor inlet temperature. Advanced ceramic composites are projected to be capable of operating up to 1649°C (3000°F) turbine rotor inlet temperature at up to 670 M/Sec (2200 ft/sec) tip speed. Current uncooled turbine blade structural technology is limited to operate at less than 610 M/Sec (2000 ft/sec) and below 1204°C (2200°F) turbine rotor inlet temperature.

These projections are based on the development of advanced materials (such as ceramic composites and carbon-carbon) and anistropic design analysis technologies.

4.5.3 Propfan Gear Box Design Guidelines

For the propfan to succeed in achieving its very significant specific fuel consumption reductions (Section 4.2.3), an advanced gearbox capable of handling counter rotating propfan shafts and 12-15 turndown ratio must be developed. In addition, gearbox weight, heat transfer, volume and reliability must be improved to be a viable solution. The gearbox design requires high capacity gearing operating at temperatures beyond the capability of current lubricants and materials.

Figures 32 and 33 present the projected gearbox design guidelines as a function of input speed and horsepower capability. A 50% reduction in weight is projected for the year 2000 gearbox technology. Similar improvements are also expected in the gearbox volume. These improvements would be achieved through advanced materials such as carburized materials for gears. The missilized design would also incorporate self-contained lubrication systems. The higher operating temperature resulting from eliminating conventional oil tank, heat exchanger and oil conditioning devices would demand lubricating oil capability to 427°C (800°F). Development of silicon and fluorosilicone oils to enhance load carrying capacity will be required. These materials have a substantially more favorable viscosity to temperature relationship.

In addition to the lubricant and material development, higher load capacity tooth geometry such as Wilhaber-Novikow profiles would be utilized. These tooth forms, which transmit the gear loads through conformal, circular surfaces rather than through double convex involute surfaces, have the capacity to handle two to three times the limit loading of the conventional gears.

The unique gearing concepts, advanced materials, high temperature lubrication oils and missilized features will be required to achieve a light weight compact and reliable gearbox for year 2000 counter rotating propfans.

4.5.4 Mechanical Design Technology

Significant enhancements in mechanical design technology will be required to satisfy the conflicting goals of improved durability, advanced thermodynamic cycles (higher temperature and pressure) and increased thrust-to-weight ratios for year 2000 technology engines. Some of the key areas are seals and bearings, shaft design, high performance gearing and lubrication systems, secondary flow system design and vibration control.

Seals

Seals are one of the primary drivers to achieve high performance and durability goals of the engine. Figure 34 shows current contacting seals technology, as well as projected requirements for the higher rotational speeds and temperatures in the year 2000. Current technology is limited to maximum rubbing velocity of 122 M/Sec (400 ft/sec) at maximum temperature of 232°C (450°F). These rubbing velocity and operational temperature limits are projected to 152 M/Sec (500 ft/sec) and 343°C (650°F) respectively for the year 2000.

These improvements will be realized through advanced seal materials which offer better wear resistance/durability characteristics. Some of the candidate materials are ceramics, high temperature metals with a low thermal coefficient of expansion (to provide compatibility with shaft materials) and high temperature carbons.

Non-contacting seals technology projections are presented in Figure 35. Current technology is also superimposed for comparison. The projections show requirements for surface velocities up to 274 M/Sec (900 ft/sec) and seal operating temperatures to 649°C (1200°F). The current technology limits are 213 M/Sec (700 ft/sec) and 537°C (1000°F). Seal pressure differential requirements are 2.07×10^6 Pa (300 psi), the current limit is 1.38×10^6 Pa (200 psi). These advances in seal materials will be provided by low thermal expansion/high modulus materials such as TZM, or ceramic composites in conjunction with graphite facing. The design technology required to make these components work along with advanced materials focus is upon the small seal heights, ranging from 2.54 MM to 3.05 MM (0.100 to 0.120 inches), while incorporating ring oil cooling. Typical seals will incorporate hybrid lift geometry, providing hydrostatic as well as hydrodynamic sealing capabilities.

Bearings

Bearing technology is one of the most critical advances required to achieve high performance through high speed rotating turbomachinery. Figure 36 presents projected bearing technology requirements. Advanced bearing design capabilities will require improved fracture toughness up to 3.5 MDN, higher tensile strength and high hot hardness (RC 56-60). Bearing operating temperatures are projected to 649°C (1200°F) for metallics and 982°C (1800°F) for ceramics. In addition, three to five times improvement in rolling contact fatigue capability is required.

The bearing design life guidelines are shown in Figure 37 as a function of bearing operating temperature for ceramics and metallic materials. The operating temperature capabilities are projected to 649°C (1200°F) for metallic and 982°C (1800°F) for ceramic bearings compared to a current technology limit for metallic bearings at approximately 454°C (850°F). Ceramic bearings operating at 982°C (1800°F) will require an improvement in material brittleness, and innovative schemes for accommodating differences in thermal expansion between shafts and the inner bearing race.

These advances will be achieved through advanced bearing materials such as dual property powder metallurgy technology, carburized metallics with iron implanted surfaces, ceramics and the hybrid ceramic/metallic materials (cermets).

Shafts

Figure 38 presents projected shaft design guidelines, focusing on the first bending mode critical speed. The projections are presented for three shaft materials; INCO 718 (current technology), Ti/Borsic composite material with 50% volume fraction ratio, and Beryllium, all based on 20% critical speed margin. Maximum shaft speed capability improvement up to 2.8 times the current technology are projected for year 2000.

These improvements are realized through advanced material properties, e.g., high modulus to density ratios (E/ρ). Fiber reinforced metals and Beryllium provide modulus to density ratio improvements of 2.2 and 7.5 times the current technology (INCO 718) respectively. However, Beryllium is a toxic material and would require special care in handling and manufacturing. In addition, a 25% improvement in the shear stress capabilities is required to reduce the shaft diameter. Shaft design limits are set by the low pressure spool shaft critical speeds and the shaft torque capability.

Other Technologies

Other key design technologies are: lubrication systems, secondary flow systems and vibration control. Higher engine overall pressure ratios and temperatures will impose a very severe operational environment for the lubrication system. Improved cooling and reduced pressure losses will be required for the secondary flow systems in order to reduce cooling air requirements and improve efficiency. As the shaft size diminishes, the vibration control will impose a challenging task to the designer. These technologies will require more attention once a specific engine design and configuration are chosen and detailed design analysis is initiated.

4.6 Materials Technology Projection

Materials and manufacturing are among the critical technologies required to satisfy the future system requirements. Materials technology provides higher turbine inlet rotor temperature and higher overall pressure ratio capability, resulting in improved performance and higher thrust-to-weight ratio engines. In addition, high temperature materials eliminate cooling penalties and thus provide further fuel consumption improvements.

Materials currently used in production engines are aluminum, titanium and steel for axial and centrifugal compressor rotors. Cast aluminum (C355) is used for inlet housing and other static structure where temperatures do not exceed approximately 149°C (300°F). Combustor shells of the nickel base alloy INCO 625 require extensive cooling. INCO 718 is used both for the outer housing and the compressor shaft. Current engines also use an investment cast INCO 713 material for the turbine inlet nozzle. Integral-bladed turbine rotors are made from cast MAR-M-247 which satisfies requirements up to about 2000°F turbine rotor inlet temperature. These materials do not satisfy the need for advanced engines where higher temperatures (up to 1927°C) are desired.

Advanced engines will use composites in the front end and composite ceramics and carbon/carbon in the hot section extensively. Carbon/carbon and composite ceramics have application to compressors and housings to produce lower cost components. Table 11 summarizes candidate materials which will satisfy the future technology requirements for the advanced turbine engines.

Inlet Ducts and Stators

Table 11 illustrates the materials and manufacturing methods for inlet ducts and stators operating in the temperature range from room temperature to 1038°C (1900°F). Carbon/polymer composites will be used extensively for both static and rotating components in the temperature range of 316-427°C (600° - 800°F), depending on the polymer development status at that time. High volume production methods of injection or transfer molding will be utilized. Powder metal titanium aluminide has been demonstrated, but needs development for net shape fabrication of components. Higher strengths in the titanium aluminide will be provided by filament reinforcement. Again, hot isostatic pressing methods for net shapes are required to provide low cost production capabilities. The use of carbon/carbon in the temperature range of 482-1038°C (900° - 1900°F) is feasible, but manufacturing methods of filament winding should be explored. The coating system for carbon/carbon operating in the 1038-1093°C (1900-2000°F) range will be significantly easier to develop than for the 1649-1927°C (3000-3500°F) capability material.

Fan/Compressor

Manufacturing technology for filament winding to provide low cost integral bladed compressor rotors using polymer composites is being explored and will provide a low cost approach. Higher temperature requirements will be satisfied with a glass ceramic, Table 11. Reinforced titanium, titanium aluminide or fiber reinforced superalloys, will also be utilized for rotating components. Low cost manufacturing techniques will be needed for these high temperature materials.

Combustor

For a combustor design to operate with little or no cooling, the use of carbon/carbon is ideal, Table 11. A coating to satisfy the 1649-1927°C (3000-3500°F) temperature will be required. Gas operating temperatures up to 1760°C (3200°F) can be realized with composite ceramics (with reduced cooling compared to a metallic material). Fiber reinforced nickel-base superalloys with a thermal barrier coating can be considered for operating temperatures of 1371-1649°C (2500° - 3000°F) with cooling.

Turbine/Nozzle

The same materials reflected for combustors will have application for turbine inlet nozzles; they offer the same advantages as (uncooled) carbon/carbon and limitations of the thermal barrier coating for nickel-base alloys, Table 11. Composite ceramics (operating temperature up to 1760°C) and fiber reinforced superalloys (operating temperatures 1371-1649°C) will require reduced level of cooling relative to metallic materials.

The turbine rotor presents the greatest challenge, and correspondingly the greatest benefit in performance via higher temperature materials. In the temperature range of 1204-1649°C (2200° - 3000°F), the fiber reinforced superalloys using fibers from moly, ceramic, or carbon offer potential strength improvements. Again, the thermal barrier coating is critical for this type of application in the 1649°C (3000°F) temperature range. Composites, ceramics and carbon/carbon fall in the same temperature range (1649-2205°C) and have the same technical requirements as stated previously.

Nozzles and Ducts

Materials and manufacturing technology requirements for the exhaust jet nozzles and ducts operating in a high temperature environment are similar to the turbine and turbine nozzles, Table 11.

Bearings

High DN bearing materials such as ceramics and ceramic composites will provide high temperature capability, Table 11. In addition, dry lubricants for temperatures up to 1093°C (2000°F) will be necessary to provide low cost functional bearings. Refer to section 4.5.4 for bearing design guidelines and limitations.

Shafts

Fiber reinforced metallics (such as titanium/borsic) and beryllium will be used for operating temperatures up to 427°C (800°F). For shafting in the 982-1093°C (1800° - 2000°F) operating range, materials such as reinforced titanium aluminide, reinforced ceramic and carbon/carbon are candidates, Table 11. Manufacturing developments will include extrusion or hip for the reinforced titanium aluminide with net or near net shape densification for the composite ceramic. The utilization of filament winding, with an appropriate coating system for the 1093°C (2000°F) range, will be necessary.

Seals

The candidate materials for the seals are ceramics, high temperature carbon, TZM and ceramic composites depending upon the operating environmental temperature, Table 11. The advanced material requirements for seals are low thermal expansion and high modulus. Graphite facing on the seals may be required.

Gearbox

Fiber reinforced metallics such as titanium aluminide will be used for the gearbox casings, Table 11. However, polymer composites with increased thermal conductivity may be used for gearbox casings for weight reduction. The manufacturing technology will be needed for filament winding or compression molding for cost reduction. Carburizing materials such as Carpenter EX53, Vasco 2 and CBS 600, may be used for gear teeth, Table 11.

The advanced materials and manufacturing methods will pace the turbine technology for the year 2000. The composites offer the biggest gains; high temperature composites such as ceramic and carbon/carbon offer the biggest payoffs and also the highest risks. Reliable, high temperature thermal barrier coatings will be essential for use on metals. Last, but not least, new NDE techniques must be developed to fully utilize composite engine materials.

4.7 Recuperator Analysis

The recuperated turbine engine provides fuel consumption benefits. However, the penalty for the incremented weight, volume and cost must be analyzed in order to select an optimum engine for a specified mission. The objective of this sub-study was to quantify these benefits/penalties for the recuperated engine cycle for the subsonic strategic cruise missile mission.

Cycle Parametric Analysis

A parametric study was conducted for a two-spool propfan cycle to evaluate the effects of recuperator effectiveness, recuperator pressure loss and overall engine pressure ratio. The study was conducted at 1649°C (3000°F) turbine rotor inlet temperature at the sea level, Mach 0.7 condition. Parametric cycle performance is presented in Figures 39 and 40 for 6 and 12% recuperator pressure losses respectively. Superimposed on each figure is the non-recuperated cycle data for direct comparison.

The results show 7.5 to 12% SFC improvements for the 0.8 effectiveness recuperator relative to the optimum non-recuperated engine cycle. These improvements in SFC are diminished to 1.5 to 6% for 0.6 recuperator effectiveness. The corresponding thrust losses for recuperator effectiveness of 0.8 and 0.6 are 16-19% and 8-13.5% respectively.

Recuperator Design

Based on the parametric engine cycle analysis, a recuperated engine cycle (Table 12) was selected as a design basis. Recuperator design parameters are summarized in Table 13. An annular type recuperator design was selected as it provides minimum manifold volume requirements, high heat transfer surface density and excellent flow distribution with minimum pressure loss, Figure 41. The recuperator, as depicted in Figure 41, was designed using plate-fin construction, with construction based on silicon carbide.

Recuperator Benefits/Penalties

Recuperated and non-recuperated engine weights and volumes were estimated for a constant thrust engine. For the selected mission, study results indicate the following:

i) SFC improvements: 4-9%
ii) Thrust losses: 13-19%
iii) Total engine weight increase: up to 37%
iv) Engine volume increase: up to 200%
v) Engine cost increase: 20-50%

Cruise missiles are, by nature, volume limited. Any increase in engine diameter due to the added recuperator volume will have serious installation implications.

This study indicates that a recuperated engine is not a cost effective solution for a subsonic strategic cruise missile application. Cost, engine complexity, weight and volume outweigh the SFC benefits. In addition, other factors such as incremental development cost, schedule and propulsion system reliability should also be considered.

However, the recuperated engine could be a very attractive candidate for a manned application such as rotorcraft where fuel savings in repeated missions accumulate significantly over the life of the system, and the installation is not critically dependent on the engine diameter.

The recuperated engine was not selected for the system (mission and life cycle cost) analysis.

4.8 Cooled Turbine Analysis

A sub-study was conducted to evaluate the effect on the engine cycle of cooling the HP turbine blades . The following two-spool engine cycle was selected to study the cooled turbine performance:

Bypass Ratio	6.0
Turbine Rotor Inlet Temperature	1371 (2500°F)
Overall Pressure Ratio	26:1

Turbine cooling assumptions used in the study are:

HP Turbine Nozzle Cooling Bleed (%)	8.2
HP Turbine Blade Cooling Bleed (%)	6.0
HP Turbine Efficiency Degradation (Pts.)	4.5
LP Turbine Nozzle Cooling Bleed (%)	3.6
LP Turbine Blade Cooling Bleed (%)	1.0
LPT Turbine Efficiency Degradation (Pts.)	0.5
Miscellaneous Cooling Bleed (%)	1.0

These assumptions are based on vane and blade metal temperatures of 1038°C and 954°C (1900°F and 1750°F), respectively. Bulk cooling effectiveness is 0.54.

The study results showed 8.9% degradation in SFC and 18.1% thrust loss at sea level, Mach 0.7. These performance losses would increase for a higher turbine rotor inlet temperature thus resulting in further performance deterioration. In addition to the performance degradation, manufacturability of small turbine blades (0.4 in height) would be a difficult task. Increased cost and complexity and durability/reliability implications are negative factors for an unmanned application. The engine performance loss with the cooled turbine shows a significant impact on the system life cycle cost, Section 5.0.

4.9 Updated Engine Cycle Analysis

Since the initial engine cycle parametric analysis was conducted using the preliminary engine component performance projections, the selected engine cycles were updated using the detailed component projections described in Sections 4.4 through 4.6. The updated cycles included fan/compressor, combustor, and turbine performance projections for the year 2000 technology. The updated cycle performance is compared with the initial cycles in Figures 42 through 48. The results indicate a good correlation between the initial and the updated cycles. Therefore, the engine performance trends indicated by the preliminary cycle analysis are valid and were used to define the thermodynamic parameters for the candidate engines to be used for the system performance analysis.

The component efficiency levels are dependent upon the component configurations (such as axial and centrifugal) and number of stages (i.e., loading requirements). Therefore, the indivdual component configurations were integrated into the final engine cycle analysis as described in Section 4.10.

4.10 Engine/Component Configuration Evaluation

From the sixty component configurations evaluated to satisfy the engine performance goals the high payoff component configurations which correspond to the updated cycles shown in Figures 42 through 48 are summarized in Table 14. Competitive engine component configurations were evaluated for common performance and loading requirements. As an example, three high pressure compressor configurations (two axials plus centrifugal, two centrifugals and one mixed flow plus one centrifugal) were designed for common pressure ratio and airflow requirements for a two spool turbofan engine, Table 14. This approach provided a direct performance and size comparison among the component configurations.

The candidate engine configurations were selected based on the following considerations:

i) A preliminary mission analysis was conducted to evaluate the relative impact of thermodynamic parameters on the vehicle launch weight. The criterion for selection was minimum vehicle launch weight.

ii) The trade factors defined in Section 3.2 indicated that the most significant performance parameter was specific fuel consumption (SFC). Therefore, lowest SFC cycles were given primary consideration.

iii) Simplicity, manufacturability and cost were also considered. For example, centrifugal/radial components were preferred over axial stages, wherever possible, due to the small engine size, and their high aero load capacity. The total number of rotating stages was kept to a minimum for cost considerations.

The selected configurations from Table 14 are:

ENGINES	Fan/LP Comp.	IP Comp.	HP Comp.	HP Turb.	IP Turb.	LP Turb.
2-Spool Turbofan (PD2 TF2)	1-Axial	--	2-Centrif	1-Radial	--	2-Axial
3-Spool Turbofan (PD3 TF9)	1-Axial	1-Axial + 1-Centrif	1-Centrif.	1-Radial	1-Radial	3-Axial
2-Spool Propfan (PD2 PF5)	1-Axial	--	2-Centrif.	1-Radial	--	2-Axial
3-Spool Propfan (PD3 PF1)	1-Axial	1-Axial + 1-Centrif	1-Centrif.	1-Radial	1-Radial	3-Axial

Engine cross sections for the two spool turbofan, three spool turbofan, and two spool propfan are presented in Figures 49 to 51. The three spool propfan cross section was not prepared because of the similiarity to the three spool turbofan configuration.

4.11 Candidate Engines

The performance and main cycle characteristics of candidate engines are summarized in Table 15, and compared to the baseline engine. This table depicts the thrust size of the engines required to satisfy the 7408 KM (4000 NM) mission range, with the exception of the baseline engine (maximum attainable range with the baseline engine is 5260 KM). Detailed engine cycle parameters are listed in Table 16 for each candidate engine. These four selected candidate engine designs were used to compute system performance, Section 5.0.

5.0 SYSTEM PERFORMANCE EVALUATION

System performance was evaluated via mission and life cycle cost (LCC) analyses of the candidate engines defined in Section 4.11. The system performance payoffs for the advanced engines were quantified and compared with the state-of-the-art baseline engine. Mission analysis provided the definition of vehicle size, engine size and fuel burned for a 7408 KM (4000 NM) range for each candidate engine and the baseline engine. These data were used to conduct system LCC analysis.

5.1 Mission Analysis

Mission analysis was conducted for the baseline and the four advanced technology candidate engines using an in-house mission analysis computer program. As outlined in Figure 52, this is an interactive program integrating the defined mission, vehicle characteristics and engine performance. This program provides the capability to compute system weight and fuel burned for each engine. The air vehicle and engines were scaled to satisfy the prescribed 7408 KM (4000 NM) range requirement, Sections 3.3 and 3.4. The analysis was based on the methodology, assumptions and ground rules defined in Section 3.0.

Table 17 presents the summary results of the mission analysis. The baseline engine did not meet the 7408 KM (4000 NM) range requirement. Therefore, the mission and LCC analyses were conducted for a 4448 N (1000 lbs) thrust (at sea level, Mach 0.7) baseline engine which achieved only 71% (5260 KM) of the desired range, however, all advanced candidate engines satisfied the 7408 KM (4000 NM) range requirements. Specific fuel consumption (SFC) has a very dominant impact on the vehicle size:

- o Vehicle launch weight and engine size decrease as the SFC is reduced.
- o The three spool propfan engine, the lowest SFC, provides the smallest launch weight vehicle.
- o Vehicle weight reduction of up to 47% is achieved with the advanced engines relative to the baseline engine.

This implies that more missiles can be carried on a launch aircraft as indicated below:

Criteria (Based On)	Increased Missile Carrying Potential Relative To Baseline Engine
Payload	47%
Missile Diameter	26%

These very significant payoffs translate into a need for fewer launch aircraft and increased firepower.

This study indicates that the engine weight and volume are of secondary importance for the long range subsonic mission with advanced technology engines. Engine volume is important to the engine/air vehicle installation: large engine volume can reduce available fuel volume in some circumstances for a constant volume vehicle.

Detailed mission performance characteristics are shown in Figures 53 through 57 for the five engines (baseline and four advanced technology). Altitude cruise requires power ranging from 100% to approximately 75%. Advanced technology turbofans require less than 54% power at sea level cruise condition, Figures 54 and 55. The excess power is used for high 'g' maneuvers in terrain following or on a hot day. The propfan uses relatively higher power settings at sea level cruise.

It should be noted that propfan core flow is less than 5% of the high bypass ratio advanced engines. The low flow provides an added benefit of reduced inlet size that will help reduce observables. Additionally, inlet volume constraints are relieved, allowing an improved inlet recovery and reduced distortion configuration.

5.2 Life Cycle Cost Analysis

System life cycle cost (LCC) analysis was conducted for the baseline and the four advanced technology candidate engines. LCC methodology and assumptions are described in Section 3.2. System LCC consists of four elements: Operational and Support (O&S), engine development, engine acquisition and vehicle acquisition. Engine development and production costs were computed using the RCA PRICE H model. Vehicle production costs were also estimated with the PRICE model. The RCA PRICE model was calibrated with current production engine and vehicle cost data to provide consistent and reliable cost estimates. Engine maintenance costs were estimated through the use of the Teledyne CAE APSICOST MOSC model.

Figure 58 summarizes relative LCC data for the four advanced technology engines. Combined engine O&S, development and acquisition costs are approximately 10% of the total system LCC for the advanced technology engines: approximately a 50% reduction relative to the baseline engine. However, advanced engines have a very significant leverage on the System LCC resulting primarily from vehicle launch weight reductions for advanced engines (section 5.1). LCC reductions of up to 41% are achieved with the advanced technology engines relative to the baseline state-of-the-art engine, Figure 58.

Additionally, increased missile carrying capacity (26-47%) of the launch aircraft (Section 5.1) would result in further cost savings if the launch aircraft costs had been included in LCC. It should be noted that the baseline engine did not satisfy the 7408 KM (4000 NM) range. Therefore, advanced turbine engine technology not only provides significant reduction in LCC but also enhances mission capability.

LCC sensitivity analysis was also conducted for the component efficiency levels, cooled turbine and fuel costs, Figure 59. The study indicates approximately a 1% increase in system LCC for one point degradation in any component efficiency. The cooled turbine provided a very large (15%) increase in LCC. This indicates the significance of uncooled turbines and the need for advanced materials. System LCC was insensitive to the fuel costs, Figure 59. It was concluded from the study that even if individual technology projections were off by three points in each component, the LCC savings would still be 29%. Therefore, only partial achievement of the proposed program payoffs would present a very high benefit/cost ratio.

6.0 TECHNOLOGY IDENTIFICATION AND PROGRAM PLANS

The SECT study resulted in a matrix of critical turbine engine technologies that were weighed and ranked based on their benefits and risks. Technology plans were generated for each identified critical technology necessary to achieve year 2000 system mission goals. The technology plans include overall schedules and milestones.

6.1 Technology Identification And Rankings

A matrix approach was used to systematically identify and summarize technology benefit/risk in meeting the year 2000 technology goals. The benefits of the technologies evaluated were ranked on a scale from 1 to 5, Table 18. Rank 1 is the least important and rank 5 is mandatory to the achievement of the year 2000 objectives. A qualitative logic statement is associated with each technology rank. This statement indicates the significance of each ranking. A qualitative risk assessment is also associated with each technology rank.

Rank 1 technologies are mixer nozzle, fuel control, propfans and composite cold static structure, Table 18. These are available or readily projectable technologies and have little risk associated with them. Multi-stage low pressure turbines and fan/LP compressors are rank 2 technologies. These do not require high technology development programs and will result from currently ongoing Research and programs. Performance and structural goals can be achieved during the component programs. Low observable coatings and materials and instrumentation are important technologies (rank 3). Technology development programs are required, however, these are considered outside the scope of the SECT study. These technologies have moderate risk associated with them.

The double centrifugal compressor, axial-centrifugal compressor, slinger combustor and high speed shafts are ranked as very important technologies (rank 4), Table 18. These technologies require programs to meet the year 2000 mission objectives, and are deemed high risk. A recommended double centrifugal compressor generic technology program encompassing the need of all the configurations is presented in Table 19.

A ceramic composite radial turbine capable of operating at 1649°C (3000°F) turbine inlet gas temperature, bearings and seals capable of operating at 649°C (1200°F) environmental temperature and a light weight missilized propfan gearbox are classed as mandatory (rank 5) technologies. These technologies are required if year 2000 mission requirements are to be met. Very high risks are associated with the very high payoff of these technologies.

6.2 SECT Program Plans

Critical technology plans have been prepared for the critical technologies (ranks 4 and 5, section 6.1) necessary to achieve projected year 2000 system objectives. Major elements of the plans are aerodynamic technology, structural/materials technology, test requirements and schedule. The technology plans are generic in nature, as they cover a wide range of component size and operational environment. A more detailed engine concept definition should preceed the technology phase to provide specific technology guidelines and requirements.

The technology plans present major milestones covering a time frame from 1986 through 1993. Engine demonstrator tests are not included, as they may be outside the NASA charter. However, these tests should be conducted in the 1993-97 time frame to verify and demonstrate the technology in a real operational environment. They can be conducted on any available size engine. Given the type of proposed programs and a follow on engine demonstrator program, mission-specific prototype testing could be scheduled in the 1997 to 2000 time frame.

Radial Inflow Turbine

Figure 60 presents the ceramic radial inflow turbine plans. Improved aerodynamics achieved through improved design methodology and high temperature ceramic material development are two key turbine technologies required to satisfy the year 2000 cruise missile mission requirements (Section 4.4.2.1). These technologies are planned as parallel efforts and brought together in integrated high performance and high temperature verification tests. These tests will be conducted in a simulated operational environment.

Bearings and Seals

Bearings and seals program plans are shown in Figure 61. Materials programs are required to meet the high speed and high temperature operating environment for seals and bearings (Section 4.5.4). High temperature lubrication bench development will be conducted in parallel with the bearing materials and integrated in bearing rig verification tests. These tests will be conducted in a simulated engine operational environment.

Propfan Gearbox

The gearbox plan is presented in Figure 62. The key technologies for the advanced missilized propfan gearbox are methodology to analyze and design the advanced tooth profiles, materials, heat rejection and high temperature lubrication (Section 4.5.3). These technologies can be evolved in parallel and verified separately. The integration of the technologies will be accomplished in full scale simulated verification tests, for example on a back-to-back rig.

Double Centrifugal Compressor

Double centrifugal technology plans are presented in Figure 63. Advanced aerodynamic methodology is required to design compressor subcomponents (individual centrifugal stages, diffusers and ducts), Section 4.4.1.1. Aerodynamic performance tests will be conducted, without advanced materials, for each centrifugal stage; these tests will be followed by overall compressor performance verification tests. Materials technology can be developed in parallel to the aerodynamic technology. Materials and aerodynamic technologies will be integrated in simulated verification tests.

Slinger Combustor

Figure 64 shows the slinger combustor technology plan. Materials and aerothermodynamic technologies (Section 4.4.3.1) can be evolved in parallel. Aerothermodynamic technology and code verification, for example, can be developed with the current (low temperature) materials. Similarly, materials/structural technology can be evolved independent of aerothermodynamic technology considerations. These technologies will be brought together in integrated full scale combustor rig verification tests.

High Speed Shafts

High speed shaft technology plans are shown in Figure 65. High modulus to density ratio materials and structural design methodology are the key to the success of high speed shaft designs in the year 2000 (Section 4.5.4). Materials and structural technologies can be evolved in parallel and integrated into shafting verification tests.

7.0 TECHNOLOGY TRANSFER TO OTHER APPLICATIONS

Teledyne CAE concentrated the SECT study effort on the advanced strategic subsonic cruise missile engine. However, advanced aerodynamic, structural and materials technologies were evaluated and reviewed to insure applicability of these technologies to rotorcraft, tactical cruise missile and APU's. Where deemed necessary, scaling can be used to meet the other three application needs from a common projected technology base. A key element to effect technology transfer is identifying the technology requirements for propulsion applications of diverse nature. This is mandatory because of peculiarities/uniqueness of small engines and the lack of broad-range exploratory funding for components and materials.

These applications share common small engine size-related design problems (flow range from 0.2 to 3.5 lbs/sec), such as maintaining manufacturing tolerances and airfoil shapes while achieving required structural integrity and performance.

Table 20 presents the major system and environmental priorities for each of the four selected application areas, and their impact on engine requirements. The Table shows, for example, both the strategic cruise missile engine and rotorcraft engine cycles are driven by a need for maximum practical pressure ratio and turbine inlet temperature, at high component efficiency, and each probably on 2 (Concentric) or 3 (Excentric) shafts. These factors define common compressor and turbine technology requirements. Differences are expected to exist in the rating of the components, and possibly some material choices. The strategic cruise missile engine will be operated at increased temperature and speed, consistent with its 50-hour life requirement. With suitable derating, similar components will provide the multi-thousand hour durability needs of the rotorcraft turboshaft. Reliability requirements are high for each, but are focused on different definitions (e.g., 99+% one-way mission achievement reliability versus high mean time between unscheduled maintenance actions). Table 20 also shows that acquisition cost is a high priority for each application. Each has a major influence on overall vehicle costs by virtue of its leverage on the system performance and overall life cycle cost.

Component/engine performance is not the primary technology requirement for the APU's and tactical cruise missile engines. However, performance improvements are desirable, especially for the tactical cruise missile system. With suitable technology derating, cost, structural integrity, distortion tolerance and size (weight and volume) objectives can be met for these applications. Low cost for the tactical cruise missile and APU's, for example, can be achieved with simplified high pressure ratio (per stage) rotating components with advanced materials and manufacturing technologies from the subsonic strategic cruise missile application. The size, configuration and general performance needs of rotorcraft engines and elements of tactical cruise missile engines and APU's are congruent with those of the strategic cruise missile engines.

Based on the evaluation of mission requirements, technology drivers and critical parameter requirements, the common technologies have been identified for the four diverse applications, Table 20. Implied in these technologies are aerodynamic and structural tool development technologies such as computational fluid mechanics technology, optimum structural designs and analysis and materials architecture technology. These are generic technologies that can be used to design, analyse and predict any turbine engine component. The subsonic strategic cruise missile engine technologies far exceed the levels required by rotorcraft, APU's and tactical missiles. Once these technologies are verified in strategic cruise missile engine environments, they will be available for other applications via technology transfer. The common technologies have varying benefits on each system, however, the unifying theme is that a combination of these technologies will benefit all candidate systems.

8.0 TECHNOLOGY PAYOFFS AND CONCLUSIONS

This study has indicated that considerable advances in future small gas turbine engines can be achieved with well planned and concentrated research activities. The system payoffs resulting from these engine technologies are very attractive and must be pursued. The following summarizes the major advanced turbine engine technology payoffs for cruise missile engines and the conclusions of this study:

(i) Significant SFC improvements (30-50%) are achieved in turbofan and propfan configurations, relative to the state-of-the-art engine.

(ii) The advanced engines provide a potential of very large increases of range relative to current operational vehicles. Further range increases can be realized with increased missile size while keeping the engine thrust below 4448 N (1000 lbs).

(iii) Reduction in missile weight and size increases the launch vehicle missile carrying capability up to 47% based on missile weight and 26% based on the missile diameter relative to the baseline engine.

(iv) System life cycle cost savings of up to 41% can be realized with the advanced technology engines.

(v) System life cycle cost is found to be relatively insensitive to fuel cost. However, sensitivity analysis indicated that even if individual efficiency levels were estimated high by three points in each component, the LCC savings would still be 29%.

(vi) High strength, high temperature composite and ceramic-composite materials technology is a major key in achieving the desired goals for the year 2000 systems. Ceramic composite radial inflow turbine technology capable of operating at 1649°C (3000°F) turbine inlet temperature is mandatory for achieving high performance at low cost.

(vii) Propfan gearbox technology is mandatory for achieving low SFC engines with a suitable installation on the air vehicle.

(viii) To fully achieve the desired small turbine engine goals, technology programs are required in all disciplines: aerodynamics, structures, materials. Instrumentation to calibrate design systems and evaluate performance must also be provided for the reduced size and higher speed flowpaths.

9.0 RECOMMENDATIONS

Significant advances in small gas turbine engines can be achieved by systematically identifying and developing technology in each technical discipline. It is therefore recommended that:

(i) NASA conduct a detailed system study with an airframer as the prime contractor to substantiate the system payoffs.

(ii) NASA conduct a more focused design concept study on the propfan e.g., to delete the axial stage and the through shaft, and thus minimize bearing development and other structures-related problems.

(iii) NASA initiate the following programs of critical technologies:

- Radial inflow turbine
- Bearings and seals
- Propfan gearbox
- Double centrifugal compressor
- Slinger combustor
- High speed shafts

(iv) If all critical technology programs can not be undertaken due to budgetary constraints, the two key enabling technologies, high temperature capability ceramic composite radial inflow turbine and propfan gearbox, should be initiated.

REFERENCE

1. M. L. Spearman, "Aerodynamic Characteristics Of A Swept-Wing Cruise Missile At Mach Numbers From 0.50 to 2.86", NASA Technical Note D-7069, November 1972.

2. CC. Ciepluch "Energy Efficient Engine - Component Development and Integration Semi-Annual Report Number 8", NASA CR-169496, April 1982.

3. R.D. Thulin, D.C. Howe and I.D. Singer, "Energy Efficient Engine High Pressure Turbine Detailed Design Report", NASA CR-165608, January 1982.

TABLE 1: BASELINE ENGINE PERFORMANCE

Alt. KM (ft)	0	0	10.7 (35,000)
Mach No.	0.7	0	0.5
Inlet Recovery	1.0	1.0	1.0
Fan Pressure Ratio (hub/tip)	2.0/2.039	2.01/2.16	2.08/2.32
Fan Efficiency (hub/tip)	85.5/81.6	84.3/80.8	78.9/74.2
Fan Corrected Airflow, Kg/sec (lb/sec)	6.00 (13.228)	6.17 (13.596)	6.39 (14.092)
Fan Corrected Speed, Rev/sec (RPM)	599 (35,978)	617 (37,023)	590 (35,438)
Bypass Ratio	1.11	1.11	1.11
HP Compressor Pressure Ratio	6.31	6.68	7.17
HP Compressor Efficiency	0.795	0.777	0.735
HP Compressor Corrected Airflow, Kg/sec (lb/sec)	1.59 (3.510)	1.63 (3.599)	1.66 (3.654)
HP Compressor Corrected Speed Rev/sec (RPM)	781 (46,849)	816 (48,947)	853 (51,160)
Combustor Pressure Loss (%)	2.5	2.5	2.5
Combustor Efficiency	0.995	0.994	0.989
Fuel Lower Heating Value, KJ/Kg (Btu/lb) - JP-10	42,100 (18,100)	42,100 (18,100)	42,100 (18,100)
HP Turbine Work, $\Delta h/\theta_{cr}$, KJ/Kg (Btu/lb)	70.9 (30.5)	70.9 (30.5)	70.7 (30.4)
HP Turbine Efficiency	0.85	0.85	0.85
HP Turbine Rotor Inlet Temperature, °C (°F)	1160 (2120)	1127 (2060)	1016 (1860)
HP Turbine Corrected Speed Rev/sec (RPM)	418 (25,062)	423 (25,355)	416 (24,930)
HP Turbine Corrected Airflow, Kg/sec (lb/sec)	0.52 (1.14)	0.52 (1.14)	0.52 (1.14)
LP Turbine Work, $\Delta h/\theta_{cr}$, KJ/Kg (Btu/lb)	45.4 (19.5)	45.1 (19.4)	45.8 (19.7)
LP Turbine Efficiency	0.88	0.88	0.88
LP Turbine Inlet Temperature, °C (°F)	883 (1621)	854 (1570)	762 (1403)
LP Turbine Corrected Speed, Rev/sec (RPM)	318 (19,085)	316 (18,970)	315 (18,920)
LP Turbine Corrected Airflow, Kg/sec (lb/sec)	1.41 (3.10)	1.41 (3.1)	1.41 (3.10)
Nozzle Thrust Coefficient	0.9864	0.9814	0.981
Thrust, Newtons (lbs)	2758 (620)	3100 (697)	778 (175)
SFC Kg/HR/N (lb/hr/lb)	0.102 (1.005)	0.071 (.697)	0.087 (0.855)
Total Airflow, Kg/sec (lb/sec)	7.94 (17.51)	6.19 (13.60)	200 (4.41)

HP Compressor Stage Performance (SL/0.7 MACH):

Stage	1	2	3	4
PR	1.31	1.34	1.2	2.99
AD. EFF.	.85	.85	.82	.817
POLY EFF.	.856	.856	.825	.843
Inlet Correct Airflow, Kg/sec (LB/sec)	1.59 (3.51)	1.27 (2.8)	1.00 (2.2)	0.86 (1.89)
Exit Correct Airflow, Kg/sec (LB/sec)	1.27 (2.8)	1.0 (2.2)	0.86 (1.89)	0.34 (.75)

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TABLE 2: BASELINE ENGINE MATERIALS

MATERIAL	COMPONENT
C355 Alum	Front Frame Gear Box Fan Stators Axial Compressor Shroud
17-4 PH	Fan Rotors & Fan Stub Shaft Axial Compr. Rotors & Stators Radial Diffuser
TI 6AL-4V	Radial Compressor Shroud Fan Thrust Bearing Housing
INCO 718	Radial Compressor Rotor HP Compressor Shaft Combustor Housing
INCO 713	HP Turbine Inlet Nozzle LP Turbine Inlet Nozzle LP Turbine Rotors
INCO 625	Combustor Exhaust Duct/Mixer
MAR-M-247	Turbine Blades
Nitralloy	LP Shaft
6061 Alum	Fan Duct
300 Series ST. ST.	Tubing & Adaptors

TABLE 3: TYPICAL ENGINE CYCLES FOR COMPONENT DESIGN
SEA LEVEL/0.7 MACH

PARAMETERS	2-Spool TURBOFAN	3-Spool TURBOFAN	2-Spool PROPFAN
FAN/LP COMPRESSOR			
$W\sqrt{\theta}/\delta$, Kg/sec (LB/SEC)	16.71 (35.65)	12.78 (28.17)	1.12 (2.48)
PR	1.65	2.0	1.7
EFFICIENCY	0.899	0.896	0.881
BYPASS RATIO	6.0	6.0	---
INTERMEDIATE COMPRESSOR			
$W\sqrt{\theta}/\delta$, Kg/sec (LB/SEC)	---	1.02 (2.255)	---
PR	---	6.71	---
EFFICIENCY	---	0.835	---
HIGH PRESSURE COMPRESSOR			
$W\sqrt{\theta}/\delta$, Kg/sec (LB/SEC)	1.52 (3.356)	0.21 (0.457)	0.73 (1.60)
PR	15.758	3.354	12.94
EFFICIENCY	0.830	0.848	.821
COMBUSTOR			
PRESSURE LOSS, %	2.0	2.0	2.0
EFFICIENCY	0.998	0.998	0.998
HIGH PRESSURE TURBINE			
REF. WORK ($\Delta h/\theta cr$), KJ/Kg (BTU/LB)	93.5 (40.2)	52.6 (22.6)	74.2 (31.9)
EFFICIENCY	0.867	0.878	0.88
TRIT, °C (°F)	1371 (2500)	1649 (3000)	1649 (3000)
$W\sqrt{\theta}\epsilon/\delta$, Kg/sec (LB/SEC)	0.22 (0.480)	0.11 (0.242)	0.14 (0.305)
INTERMED. PRESSURE TURBINE			
REF. WORK ($\Delta h/\theta cr$), KJ/Kg (BTU/LB)	---	59.3 (25.5)	---
EFFICIENCY	---	0.867	---
TRIT, °C (°F)	---	1391 (2536)	---
$W\sqrt{\theta}\epsilon/\delta$, Kg/sec (LB/SEC)	---	0.22 (0.481)	---
LOW PRESSURE TURBINE			
REF. WORK ($\Delta h/\theta cr$), KJ/Kg (BTU/LB)	89.6 (38.5)	112.1 (48.2)	125.1 (53.8)
EFFICIENCY	0.913	0.901	0.894
TRIT, °C (°F)	961 (1762)	1134 (2073)	1287 (2348)
$W\sqrt{\theta}\epsilon/\delta$, Kg/sec (LB/SEC)	0.84 (1.857)	0.48 (1.067)	0.38 (0.831)
PERFORMANCE			
THRUST, Newtons (LB)	3447 (775)	3447 (775)	3336 (750)
SFC, Kg/HR/N (LB/HR/LB)	0.074 (0.722)	0.072 (0.703)	0.054 (0.526)

TABLE 4
ENGINE COMPONENT CONFIGURATION MATRIX

COMPONENT	TURBOFANS		PROPFANS	
	TWO-SPOOL	THREE-SPOOL	TWO-SPOOL	THREE-SPOOL
FAN/LP COMP.	1A	1A	1A	1A
INTER. PRESS. COMP.	—	1C, 1A + 1C	—	1-C
HP COMP.	2A + 1C, 3A + 1C MF + C, C + C	1C	2A + 1C, 3A + 1C MF + C, C + C	1C
HP TURBINE	1A, MF,1R	1R	1A, 1R	1R
IP TURBINE	—	1A, 1R	—	1A, 1R
LP TURBINE	3A, MF	3A, MF	3A, MF	3A, MF
COMBUSTOR	SLINGER	ANNULAR SLINGER	SLINGER	ANNULAR

A — AXIAL
C — CENTRIFUGAL
MF — MIXED FLOW
R — RADIAL

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TABLE 5: COMPRESSOR DESIGN GUIDELINES

COMPRESSOR CONFIGURATION	MAXIMUM TEMPERATURE COEFFICIENT	MINIMUM BLADE HEIGHT, MM (IN)	BACKWARD CURVATURE
AXIAL	0.4	7.62 (0.3)	---
MIXED FLOW	0.55	5.08 (0.2)	---
CENTRIFUGAL	---	2.54 (0.1)	30°

TABLE 6: TURBINE TECHNOLOGY IMPROVEMENTS AND PAYOFFS

TECHNOLOGY	PAYOFF
o 3D AIRFOIL DESIGN	o IMPROVED VANE/BLADE OPTIMIZATION
o HIGH MACH NUMBER BLADES	o INCREASED STAGE REACTION
o 3D VISCOUS FLOW ANALYSIS	o REDUCED SECONDARY FLOW LOSSES
o VANE-BLADE INTERACTION	o IMPROVED AERO/REDUCED COOLING
o ACTIVE/PASSIVE TIP CLEARANCE CONTROL	o REDUCED LEAKAGE LOSSES
o MATERIAL IMPROVEMENTS	o HIGHER WHEEL SPEED AND AN^2
o MANUFACTURING IMPROVEMENTS	o IMPROVED AIRFOIL TOLERANCES

TABLE 7
TURBINE DESIGN CONSTRAINTS

AXIAL

- BLADE HEIGHT > 10.16 MM (0.4 in)
- POSITIVE HUB REATION
- $M_2 < 1.25$
- $R_{HUB}/R_{TIP} \geq 0.7$
- $\Delta\beta_{HUB} < 130°$
- CARBON-CARBON:

o $AN^2 < 9.96 X 10^8$ M^2-REV^2/SEC^2 (1960 X 10^8 IN^2-RPM^2)

o TIP SPEED < 914 M/Sec (3000 ft/sec)

o RADIAL ELEMENTS (Low Twist)

- CERAMICS COMPOSITES

o $AN^2 < 6.10 X 10^8$ M^2-Rev^2/Sec^2 (1200 X 10^8 IN^2-RPM^2)

o TIP SPEED < 655 M/Sec (2150 ft/sec)

MIXED FLOW

- BLADE HEIGHT > 7.62 MM (0.3 in)
- FAVORABLE REACTION ($W_2 = 2.5W_1$)
- CARBON-CARBON:

o $AN^2 < 9.96 X 10^8$ M^2-Rev^2/Sec^2 (1960 X 10^8 IN^2-RPM^2)

o TIP SPEED < 914 M/Sec (3000 ft/sec)

o RADIAL BLADE

- CERAMIC COMPOSITES

o $AN^2 < 6.1 X 10^8$ M^2-Rev^2/Sec^2 (1200 X 10^8 IN^2-RPM^2)

o TIP SPEED < 640 M/Sec (2100 ft/sec)

o 5° LEAN @ EXIT

RADIAL

- BLADE HEIGHT > 5.08 MM (0.2 in)
- FAVORABLE REACTION ($W_2 = 2W_1$)
- $D_1/D_T > 1.30$
- $D_H/D_T > 0.4$
- $R_{HUB} \geq$ 10.16 MM (0.4")
- $\alpha_1 < 80°$
- CARBON-CARBON:

o TIP SPEED < 914 M/Sec (3000 ft/sec)

o CERAMIC COMPOSITES

o TIP SPEED < 670 M/Sec (2200 ft/sec)

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TABLE 8: COMBUSTOR TECHNOLOGY PROJECTIONS

PERFORMANCE PARAMETERS	CURRENT TECHNOLOGY ASSESSMENT	TECHNOLOGY PROJECTIONS 2000
Efficiency	99.9	99.9
Pressure Loss	3-4%	$\leq$ 2%
Temperature Pattern Factor	.2 - .3	$\leq$ .15
Radial Profile Factor	As Req'd	As Req'd
Cooling Effectiveness	.6	.9

TABLE 9
STRUCTURAL DESIGN CONSTRAINTS FOR SUBSONIC STRATEGIC CRUISE MISSILE ENGINES

COMPONENT	APPLICABLE DESIGN REQUIREMENT LCF	CREEP/ FATIGUE	VIBRATION /HCF	FLUTTER	BURST/ YIELD	INGESTION /FOD	SURGE	DAMAGE TOLERANCE	MANEUVER /EXTERNAL LOADS	CONTAINMENT /BLADE OUT
FAN ROTOR	X		X	X	X *				X	
AXIAL COMP. ROTOR	X	X	X	X	X *				X	
CENT. COMP. ROTOR	X	X	X		X *				X	
FAN STATORS	X		X *	X						
COMP. STATORS	X		X *	X						
FAN CASE	X		X						X *	
COMP. HOUSING	X	X	X						X *	
FRONT FRAME	X		X						X *	

X - DESIGN REQUIREMENTS APPLIES

* - DURABILITY LIMITING FOR PROJECTED 2000 TECHNOLOGY LEVELS

TABLE 10
STRUCTURAL DESIGN REQUIREMENTS FOR CRUISE MISSILE ENGINE TURBINES

APPLICATION	TURBINE COMPONENT	STRUCTURES DESIGN REQUIREMENTS BURST	CREEP RUPTURE	LOW CYCLE FATIGUE	VIBRATION /HCF	DAMAGE TOLERANCE	MANEUVER LOADS	CONTAINMENT
LIMITED LIFE TURBINE L&S	DISK	X	*	X	X	X		
	BLADE		*	X	X			
	NOZZLE		*					
	CASE		*				*	

X DESIGN REQUIREMENTS IS APPLICABLE

* DESIGN REQUIREMENT IS A PRIME DRIVER IN ADVANCING TECHNOLOGY APPLICATIONS

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TABLE 11
ADVANCED ENGINE MATERIALS

COMPONENTS	MAX. MAT'L TEMP. CAPABILITY, °C (°F)	MATERIALS	MANUFACTURING
INLET DUCTS & STATORS	427 (800) 482 (900 1038 (1900) 1038 (1900)	C/POLYMER COMPOSITES REINFORCED CAST ALUMINUM REINFORCED P/M TI ALUMINIDE CARBON/CARBON	INJECTION/TRANSFER MOLD INVESTMENT CAST/POWDER METALS HIP FOR NET SHAPE FILAMENT WOUND (COATING FOR 1900°F)
FAN/COMPRESSOR	482 (900) 1371 (2500) 760 (1400) 1093 (2000)	POLYMER COMPOSITES CERAMIC COMPOSITES CARBON/CARBON FIBER REINFORCED SUPERALLOYS/TiAl	FILAMENT WOUND OR COMPRESSION MOLD NET SHAPE FORMING FILAMENT WOUND WITH COATING TO 2000°F P/M OR CASTING
COMBUSTOR	1649 (3000) 1760 (3200) 2204 (4000)	FIBER REINFORCED SUPERALLOYS CERAMIC COMPOSITES CARBON/CARBON	REINFORCED SHEET MANUF./FORMING/JOINING/COATING NET SHAPE FABRICATION COATING RELIABILITY NET SHAPE FABRICATION
TURBINE/NOZZLE	1649 + (3000 +) 1760 (3200) 2204 (4000)	FIBER REINFORCED SUPERALLOY/TBC CERAMIC COMPOSITES CARBON/CARBON	NET SHAPE FORMING NET SHAPE FORMING COATING RELIABILITY NET SHAPE FORMING
NOZZLES/DUCT	1649 + (3000 +) 1760 (3200) 2204 (4000)	FIBER REINFORCED SUPERALLOY/TBC CERAMIC COMPOSITES CARBON/CARBON	NET SHAPE FORMING NET SHAPE FORMING COATING RELIABILITY NET SHAPE FORMING
BEARINGS	1093 (2000)	CERAMIC	NET SHAPE FORMING
SHAFT	427 (800) 427 (800) 982 (1800) 1093 (2000) 1093 (2000)	FIBER REINFORCED METALLICS BERYLLIUM FIBER REINFORCED SUPERALLOYS REINFORCED CERAMIC CARBON/CARBON	NEAR NET SHAPE HIP NEAR NET SHAPE HIP EXTRUSION OR HIP NET SHAPE FORMING FILAMENT WOUND WITH COATING FOR 2000°F
SEALS: CONTACTING NON CONTACTING GASPATH	343 (650) 649 (1200) 982 (1800)	CERAMIC HIGH TEMP. CARBON CERAMIC, TZM, CERAMIC COMPOSITES CERAMIC, CERAMIC COMPOSITES	NET SHAPE FORMING NET SHAPE FORMING NET SHAPE FORMING
GEARBOX	482 (900) 482 (900) 482 (900)	POLYMER COMPOSITES HIGH TEMP CARBURIZED MATLS. (GEARS) FIBER REINFORCED METALLICS (CASINGS)	FILAMENT WOUND OR COMPRESSION MOLD NEAR NET SHAPE + POST SURFACE HARDENING NEAR NET SHAPE HIP

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TABLE 12
RECUPERATOR DESIGN ENGINE CYCLE: PROPFAN CYCLE

SEA LEVEL MACH 0.7	
INLET RECOVERY	0.96
LP COMPRESSOR	
Pressure Ratio	1.7
Corrected Airflow, Kg/sec (LB/SEC)	1.12 (2.48)
Efficiency	0.891
HP COMPRESSOR	
Pressure Ratio	8.82
Corrected Airflow, Kg/sec (LB/SEC)	0.73 (1.60)
Efficiency	0.823
COMBUSTOR	
Efficiency	0.998
Pressure loss, %	2.0
Fuel Heating Value, KJ/Kg (BTU/LB)	42,100 (18,100)
HP TURBINE	
Turbine Inlet Temp, °C (°F)	1482 (2700)
$\Delta H/\theta_{cr}$, KJ/Kg (BTU/LB)	65.4 (28.1)
Efficiency	0.885
LP TURBINE	
Turbine Inlet Temp, °C (°F)	1182 (2160)
$\Delta H/\theta_{cr}$, KJ/Kg (BTU/LB)	105.4 (45.3)
Efficiency	0.895
RECUPERATOR	
Effectiveness	0.7
Pressure Loss, %	6.0
PERFORMANCE	
Thrust, Newtons (LB)	2535 (570)
SFC, Kg/HR/N (LB/HR/LB)	0.051 (0.504)

TABLE 13
RECUPERATOR DESIGN PARAMETERS

Exhaust Gas In	
Temp., °C (°F)	761 (1402)
Pressure, Pa (psia)	1.33×10^5 (19.27)
Flow, Kg/sec (lb/sec)	1.47 (3.237)
Exhaust Gas Out	
Temp., °C (°F)	562 (1043)
Pressure, Pa (psia)	1.25×10^5 (18.1)
Air In	
Temp., °C (°F)	476 (889)
Pressure, Pa (psia)	20.13×10^5 (292)
Flow, Kg/sec (lb/sec)	1.43 (3.157)
Air Out	
Temp., °C (°F)	675 (1247)
Pressure, Pa (psia)	18.96×10^5 (275)
Effectiveness	.70
Total Pressure Drop $\Delta P/P$% (hot & cold sides)	12

TABLE 14
COMPONENT AND ENGINE CONFIGURATIONS

A: AXIAL
C: CENTRIFUGAL
MF: MIXED FLOW
R: RADIAL

ENGINES	TRIT °C (°F)	OPR	BPR	FAN/LP COMP.	IP COMP.	HP COMP.	HP TURB.	IP TURB.	LP TURB.
2-SPOOL TURBOFAN									
PD2TF1	1371(2500)	26	6	1A	--	2A+1C	1A	--	2A
PD2TF2	1371(2500)	26	6	1A	--	2C	1R	--	2A
PD2TF3	1371(2500)	26	6	1A	--	1MF+1C	1A	--	2A
PD2TF4	1371(2500)	26	4	1A	--	2A+1C	1A	--	2A
PD2TF5	1648(3000)	26	6	1A	--	2A+1C	1A	--	2A
PD2TF6	1648(3000)	30	6	1A	--	2A+1C	1A	--	2A
PD2TF7	1648(3000)	30	4	1A	--	2A+1C	1A	--	2A
3-SPOOL TURBOFAN									
PD3TF1	1648(3000)	45	6	1A	1C	1C	1R	1R	3A
PD3TF2	1648(3000)	45	6	1A	1A+1C	1C	1R	1R	3A
PD3TF3	1648(3000)	45	6	1A	1C	1C	1R	1R	1A+1MF
PD3TF4	1648(3000)	45	4	1A	1A+1C	1C	1R	1R	3A
PD3TF5	1648(3000)	37	6	1A	1A+1C	1C	1R	1R	3A
PD3TF6	1927(3500)	45	6	1A	1A+1C	1C	1R	1R	3A
PD3TF7	1927(3500)	45	4	1A	1A+1C	1C	1R	1R	3A
PD3TF8	1927(3500)	37	6	1A	1A+1C	1C	1R	1R	3A
PD3TF9	1371(2500)	37	6	1A	1A+1C	1C	1R	1R	3A
PD3TF10	1371(2500)	37	4	1A	1A+1C	1C	1R	1R	3A
2-SPOOL PROPFAN									
PD2PF1	1648(3000)	22	--	1A	--	2A+1C	MF	--	2A
PD2PF2	1648(3000)	22	--	1A	--	2C	1A	--	2A
PD2PF3	1648(3000)	22	--	1A	--	2A+1C	1R	--	2A
PD2PF4	1648(3000)	30	--	1A	--	2C	1R	--	2A
PD2PF5	1648(3000)	22	--	1A	--	2C	1R	--	2A
3-SPOOL PROPFAN									
PD3PF1	1648(3000)	45	--	1A	1A+1C	1C	1R	1R	3A
PD3PF2	1648(3000)	37	--	1A	1A+1C	1C	1R	1R	3A

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TABLE 15
CANDIDATE ENGINES: PERFORMANCE AT MISSION-MATCH THRUST

SEA LEVEL MACH 0.7
INLET REC. = 0.96
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ENGINE	THRUST, NEWTONS (LB)	SFC, KG/HR/N (LB/HR/LB)	BPR	TRIT, °C (°F)	OPR	NO. OF STAGES
BASELINE (2-SPOOL TURBOFAN)	4448 (1000)	0.015 (1.031)	1.1 1.1	1160 (2120)	12.6	9
2-SPOOL TURBOFAN (LOWEST SFC)	3772 (848)	0.073 (0.716)	6.0	1371 (2500)	26	6
3-SPOOL TURBOFAN (LOWEST SFC)	3278 (737)	0.069 (0.673)	6.0	1371 (2500)	37	9
2-SPOOL PROPFAN (COMPLEX/PERF. TRADE)	2153 (484)	0.054 (0.533)	— —	1649 (3000)	22	6
3-SPOOL PROPFAN (LOWEST SFC)	1993 (448)	0.051 (0.500)	—	1649 (3000)	45	9

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TABLE 16: CANDIDATE ENGINES: DETAILED CYCLE PERFORMANCE
AT MISSION-MATCH THRUST

Sea Level
Mach 0.7
Inlet Recovery 0.96

Parameters	2-Spool Turbofan	3-Spool Turbofan	2-Spool Propfan	3-Spool Propfan
Performance				
Thrust, Newtons (lb)	3772 (848)	3278 (737)	2153 (484)	1993 (448)
SFC, Kg/HR/N (lb/hr/lb)	.073 (0.716)	.069 (0.673)	.054 (0.533)	.051 (0.500)
Overall Pressure Ratio	26	37	22	45
Bypass Ratio	6.0	6.0	-	-
TRIT, °C (°F)	1371 (2500)	1371 (2500)	1649 (3000)	1649 (3000)
Airflow, Kg/Sec (lb/sec)	22.4 (49.38)	20.3 (44.67)	0.91 (2.01)	0.94 (2.08)
Fan/LP Compressor				
Pressure Ratio	1.65	1.65	1.7	2.5
Efficiency	0.917	0.913	0.838	0.885
Corrected Airflow, Kg/Sec (lb/sec)	17.61 (38.84)	15.93 (35.12)	0.72 (1.58)	0.74 (1.64)
Speed, Rev/Sec (RPM)	479 (28,735)	504 (30,220)	1660 (99,570)	-
Corrected Speed, Rev/Sec (RPM)	457 (27,420)	481 (28,850)	1584 (95,025)	-
IP Compressor				
Pressure Ratio	-	6.7	-	6.0
Efficiency	-	0.866	-	0.867
Corrected Airflow, Kg/sec (lb/sec)	-	1.50 (3.31)	-	0.35 (0.761)
Speed, Rev/Sec (RPM)	-	1101 (66,070)	-	-
Corrected Speed, Rev/Sec (RPM)	-	972 (58,340)	-	-
HP Compressor				
Pressure Ratio	15.78	3.35	12.95	3.0
Efficiency	0.824	0.861	0.838	0.839
Corrected Airflow, Kg/sec (lb/sec)	1.66 (3.65)	0.31 (0.676)	0.46 (1.017)	0.08 (0.168)
Speed, Rev/sec (RPM)	1165 (69,920)	2231(133,840)	2138(130,690)	
Corrected Speed, Rev/sec (RPM)	1029 (61,755)	1466 (87,970)	1911(114,645)	
HP Turbine				
$\Delta H/\Theta_{cr}$, KJ/Kg (Btu/lb)	95.8 (41.2)	56.8 (24.4)	72.6 (31.2)	48.6 (20.9)
Efficiency	0.882	0.883	0.868	0.866
Corrected Airflow, Kg/sec (lb/sec)	0.24 (0.53)	0.15 (0.335)	0.09 (0.19)	0.05 (0.10)
IP Turbine				
$\Delta H/\Theta_{cr}$, KJ/Kg (Btu/lb)	-	64.9 (27.9)	-	56.3 (24.2)
Efficiency	-	0.898	-	0.876
Corrected Airflow, Kg/sec (lb/sec)	-	0.32 (0.71)	-	0.08 (0.187)
LP Turbine				
$\Delta H/\Theta_{cr}$, KJ/Kg (Btu/lb)	89.3 (38.4)	95.8 (41.2)	124.4 (53.5)	133.5 (57.4)
Efficiency	0.892	0.906	0.898	0.885
Corrected Airflow, Kg/sec (lb/sec)	0.91 (2.00)	0.75 (1.65)	0.24 (0.53)	0.18 (0.39)

TABLE 17
SYSTEM PERFORMANCE SUMMARY

SL/0.7
INLET REC. = 0.96

PARAMETERS	BASELINE 2-SPOOL TURBOFAN	2-SPOOL TURBOFAN	3-SPOOL TURBOFAN	2-SPOOL PROPFAN	3-SPOOL PROPFAN
ENGINE PERFORMANCE:					
THRUST, NEWTONS (LB)	4448 (1000)	3772 (848)	3278 (737)	2153 (484)	1993 (448)
SFC, KG/HR/N (LB/HR/LB)	0.105 (1.031)	0.073 (0.716)	0.069 (0.673)	0.054 (0.533)	0.051 (0.500)
AIRFLOW, KG/SEC (LB/SEC)	12.8 (28.1)	22.4 (49.4)	20.3 (44.7)	0.91 (2.0)	0.95 (2.1)
ENGINE WEIGHT, KG (LB)	103.0 (227)	44.4 (97.9)	41.3 (91.0)	25.9 (57.2)	26.2 (57.8)
MISSILE PERFORMANCE:					
LAUNCH WEIGHT, KG (LB)	1484 (3271)	1470 (3241)	1270 (2800)	864 (1904)	782 (1725)
FUEL WEIGHT, KG (LB)	799 (1761)	853 (1880)	709 (1563)	426 (940)	367 (809)
LENGTH, M (FT)	5.9 (19.3)	5.8 (19.0)	5.4 (17.7)	4.5 (14.8)	4.4 (14.3)
DIAMETER, M (FT)	0.68 (2.23)	0.67 (2.20)	0.63 (2.05)	0.52 (1.71)	0.50 (1.65)
MIXED MISSION RANGE, KM (NM)	5260 (2840)	7408 (4000)	7408 (4000)	7408 (4000)	7408 (4000)

TABLE 18
TECHNOLOGY IDENTIFICATION AND RANKINGS

RANK/BENEFITS	PROGRAM	RISK
1 INCIDENTAL/AVAILABLE	• MIXER NOZZLE, FUEL CONTROL, PROPFANS, COMPOS. COLD STATIC	
2 NOT VERY IMPORTANT	• LP TURBINE • FAN/LP COMPRESSOR	LOW
3 IMPORTANT	• L.O. COATINGS/MATERIALS • INSTRUMENTATION	MODERATE
4 VERY IMPORTANT	• DOUBLE CENTRIFUGAL COMPRESSOR • AXI-CENTRIFUGAL COMPRESSOR • SLINGER COMBUSTOR • SHAFTS (HIGH SPEED AND HIGH LOAD)	HIGH
5 MANDATORY	• CERAMIC COMPOSITE RADIAL TURBINE (3000°F) • BEARINGS/SEALS 649°C, (1200°F) • LIGHT WT. MISSILIZED PROPFAN GEARBOX	VERY HIGH

TABLE 19

SUBSONIC STRATEGIC CRUISE MISSILE CENTRIFUGAL COMPRESSOR REQUIREMENTS

	2-SP Turbofan		3-Spool Turbofan		2-Spool Propfan (HPC)		RECOM. PROGM.	
	1-STG HPC	2-STG HPC	IPC	HPC	1-STG	2-STG	1-STG	2-STG
CORR. AIRFLOW, Kg/sec (LB/SEC)	1.661 (3.662)	0.254 (.561)	0.842 (1.856)	0.307 (.676)	0.456 (1.027)	0.090 (0.198)	0.466 (1.027)	0.090 (0.198)
PR	9.25	1.70	4.47	3.35	7.5	1.725	9.25	1.7
POLY. EFF.	.877	.867	.889	.882	.888	.895	.888	.895
AD. EFF.	.837	.857	.866	.861	.855	.853	.849	.887
Inlet MACH	.58	.30	.515	.55	.55	.30	-	-
Hub/Tip	.50	.50	.60	.50	.50	.50	-	-
Ns	85.4	83.4	86.2	90.1	92.6	93.4	-	-

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TABLE 20
SYSTEM, MISSION AND ENVIRONMENTAL PRIORITY DRIVERS:
TECHNOLOGY TRANSFER TO OTHER APPLICATIONS

APPLICATION	MISSION REQU'T	TECHNOLOGY DRIVERS	CRITICAL PARAMETERS	COMMON TECHNOLOGIES
STRATEGIC CME	• SUBSONIC MACH NO. • LIFE: UP TO 50 HRS • MAX RANGE • HIGH DISTORTION • SURVIVABILITY • LTD. VOLUME • SHOCK, EMP ETC • ALT. 0-40K FT. • STORABILITY (5 YRS. +)	• SFC • COST • Fn/VOL • Fn/Afr • OBSERVABLES • MISSION REL. • START/ACCEL.	• CORRECTED FLOW: .09-1.59 KG/SEC (0.23-3.5 LB/SEC) • OPR: 22 TO 45 • TRIT 1371-1649°C, (2500-3000°F) • COMPON. EFF'Y • POSS. MULTI-FUEL • STRUCT. INTEGRITY — HCF; S.R. — SHOCK • DISTORTION TOLERANCE • RAPID ACCEL	• CORE COMP. P.R. • HIGH TRIT/LOW COOLING LOSSES • COMPONENT EFF'Y • MULTI-FUEL COMBUSTOR • COMBUSTOR TDF CONTROL • DISTORTION TOLERANCE • FUEL CONTROL • MATERIALS • AERO STRUCT DESIGN SYSTEM • POSS. VARIABLE GEOM. COMPONENTS • MFG. METHODS • SHAFT DYNAMICS • BEARINGS & SEALS • INSTRUMENTATION
ROTORCRAFT TURBOSHAFT	• SUBSONIC MACH NO. • ALT, 0-3.0 KM (0-10K FT) • LIFE: 5000 + HRS; • MIN. FUEL BURN • OEI/EMERG. POWER • INLET SEPARATOR/ DISTORTION • SURVIVABILITY • ICING • SERVICE ACCESS. • HIGH MTBUR	• SFC • COST • OBSERVABLES • MISSION REL. • WEIGHT • START/ACCEL • DURABILITY	• OPR > 16:1 • TRIT > 1316°C, (2400°F) • COMPON. EFF'Y • ALTERN. H/C FUEL • STRUCT. INTEGRITY — HCF — LCF • RAPID ACCEL	
APU SHORT EXTENDED USAGE	• STATIC FLIGHT CONDITION • ALTITUDE 0-10K FT • LIFE: 5000 HRS • HIGH MTBUR	• COST • HP/VOLUME • WEIGHT • REL.	• STRUCTURAL INTEGRITY • RAPID ACCEL.	
TACTICAL CME	• SUB/SUPERSONIC MACH NO. • LIFE: UP TO 20 HRS • HIGH DISTORTION • SURVIVABILITY • LTD. VOLUME • ALT, 0-12.2 KM (0-40K FT) • STORABILITY	• COST • Fn/VOL • Fn/Afr • Fn/WT. • OBSERVABLES • MISSION REL. • START/ACCEL.	• POSS. MULTI-FUEL • STRUCTURAL INTEGRITY • DISTORTION TOLERANCE • RAPID ACCEL.	

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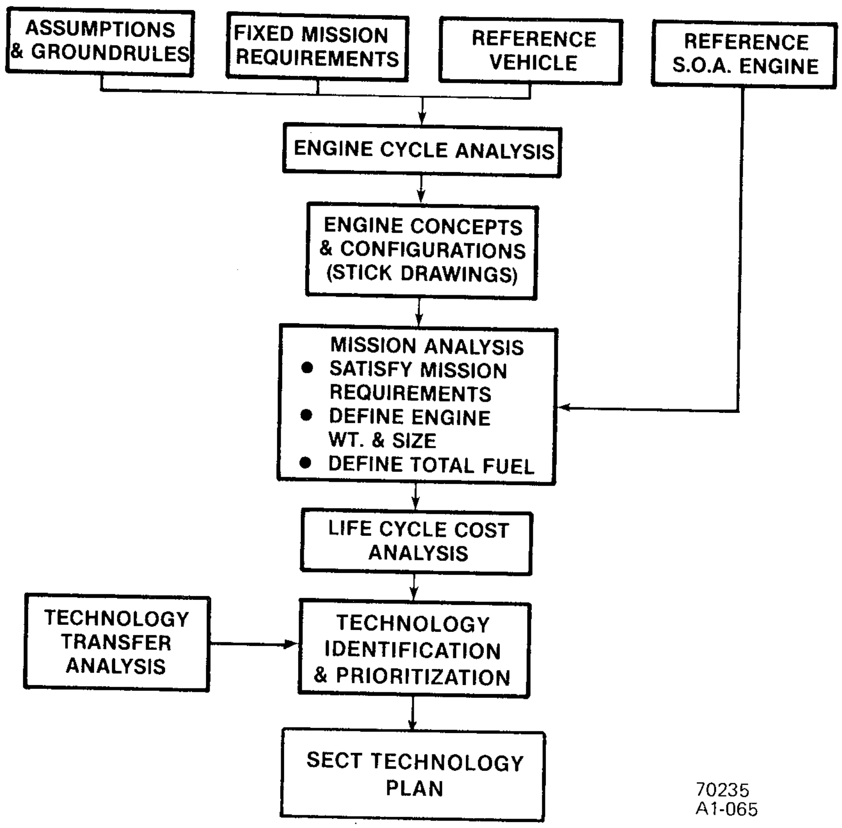

Figure 1. SECT Study Methodology Flow Chart.

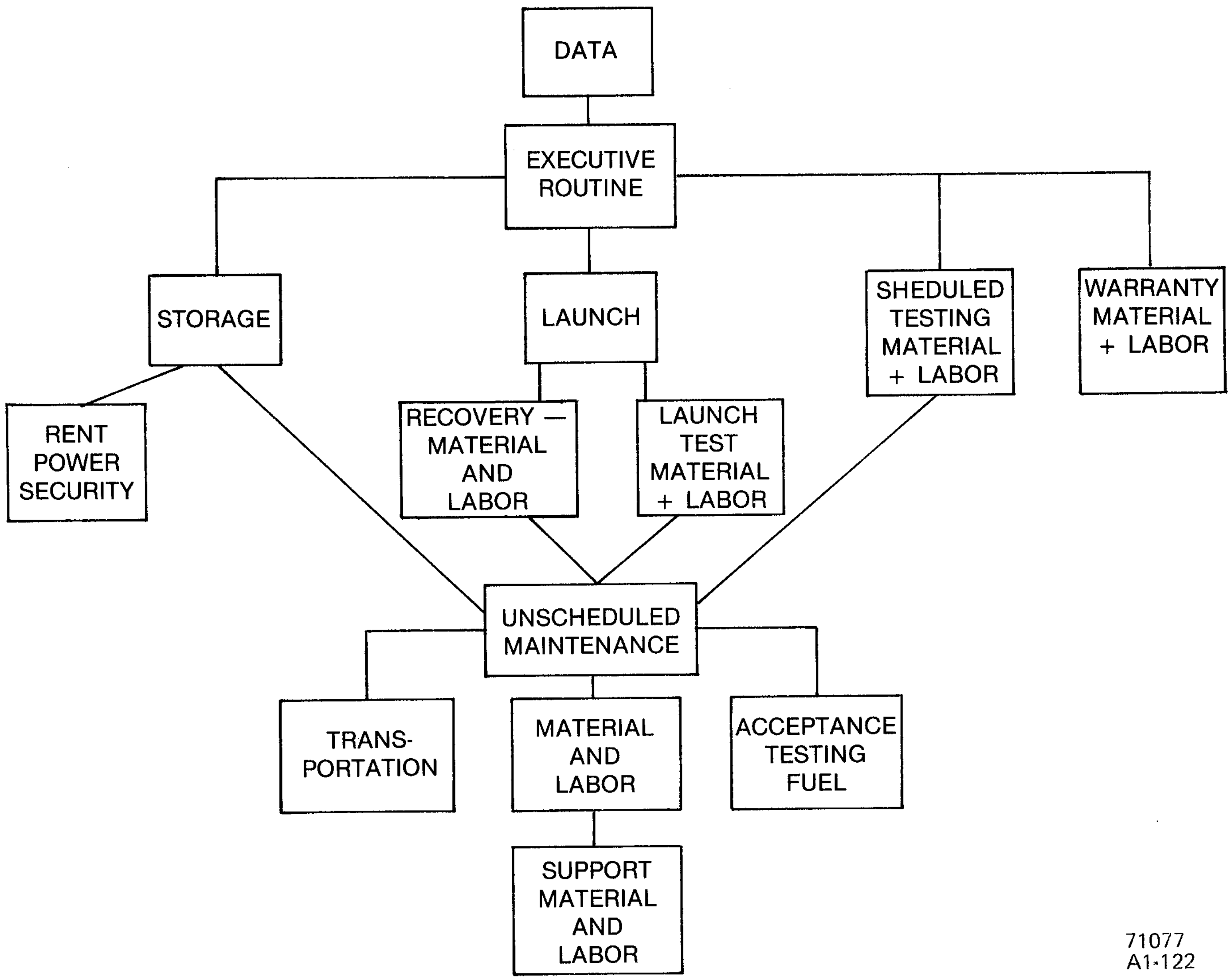

Figure 2. SECT Life Cycle Cost Study (Missile Operation and Support) Methodology.

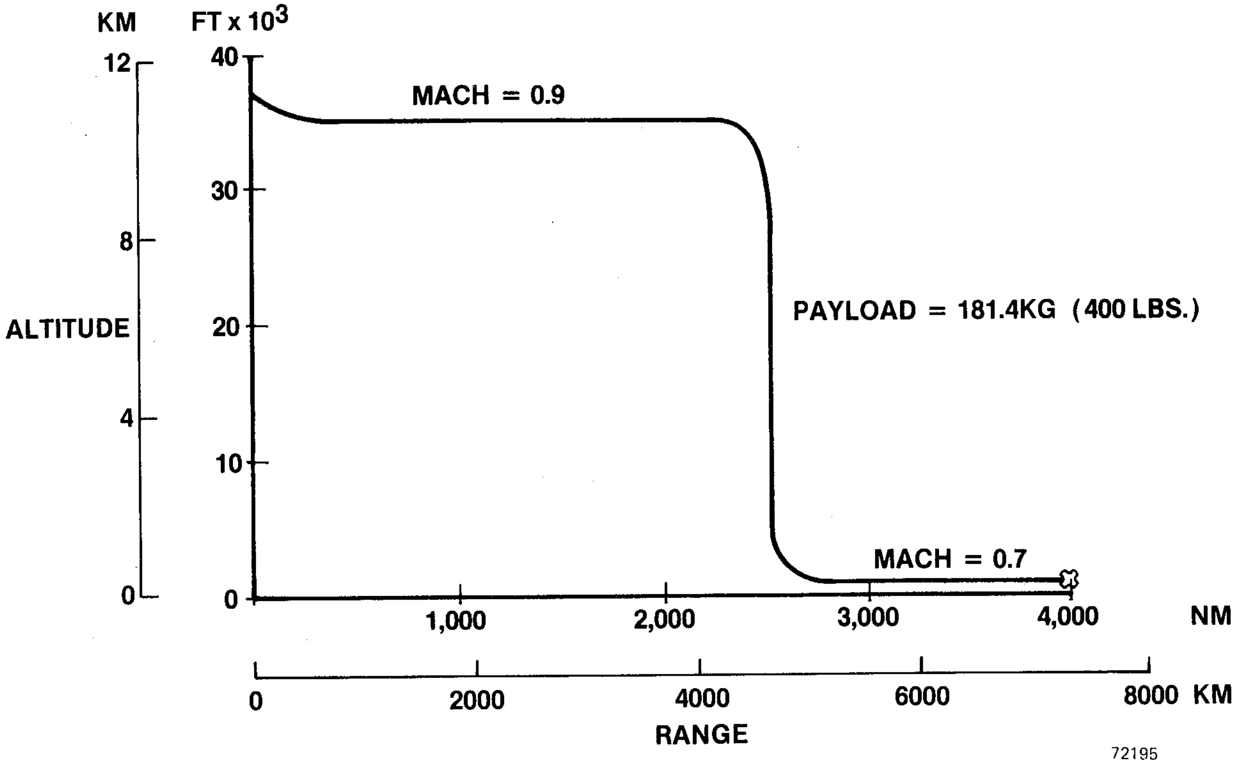

Figure 3. Basic Subsonic Strategic Cruise Missile Mission.

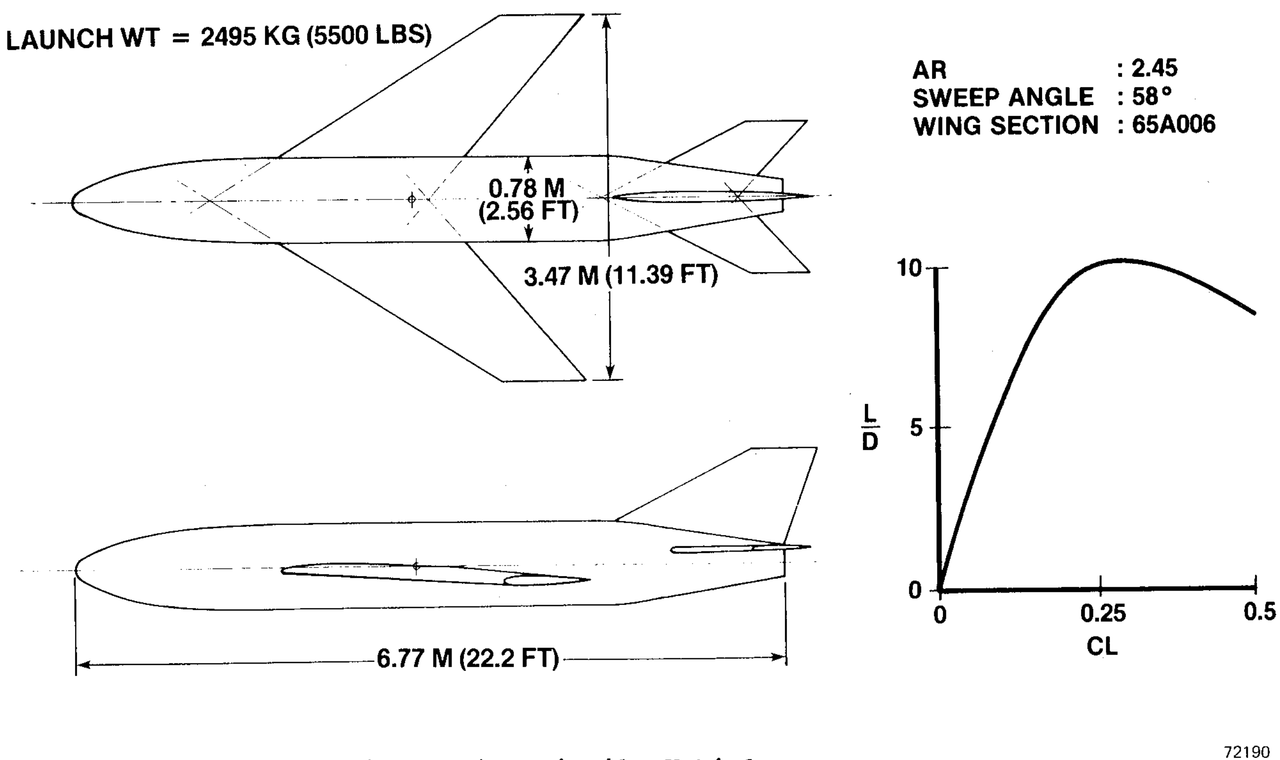

Figure 4. Baseline Cruise Missile Vehicle.

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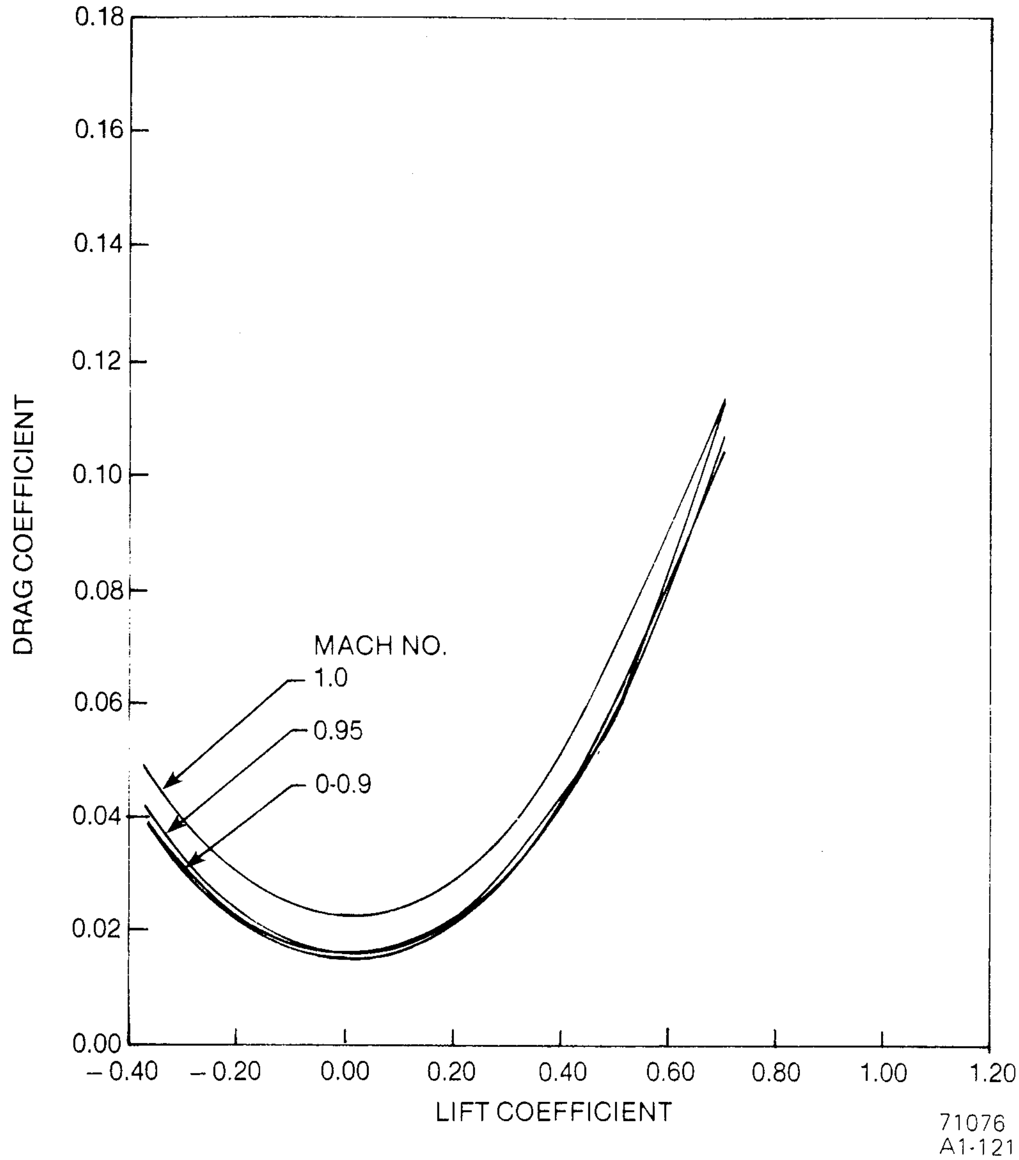
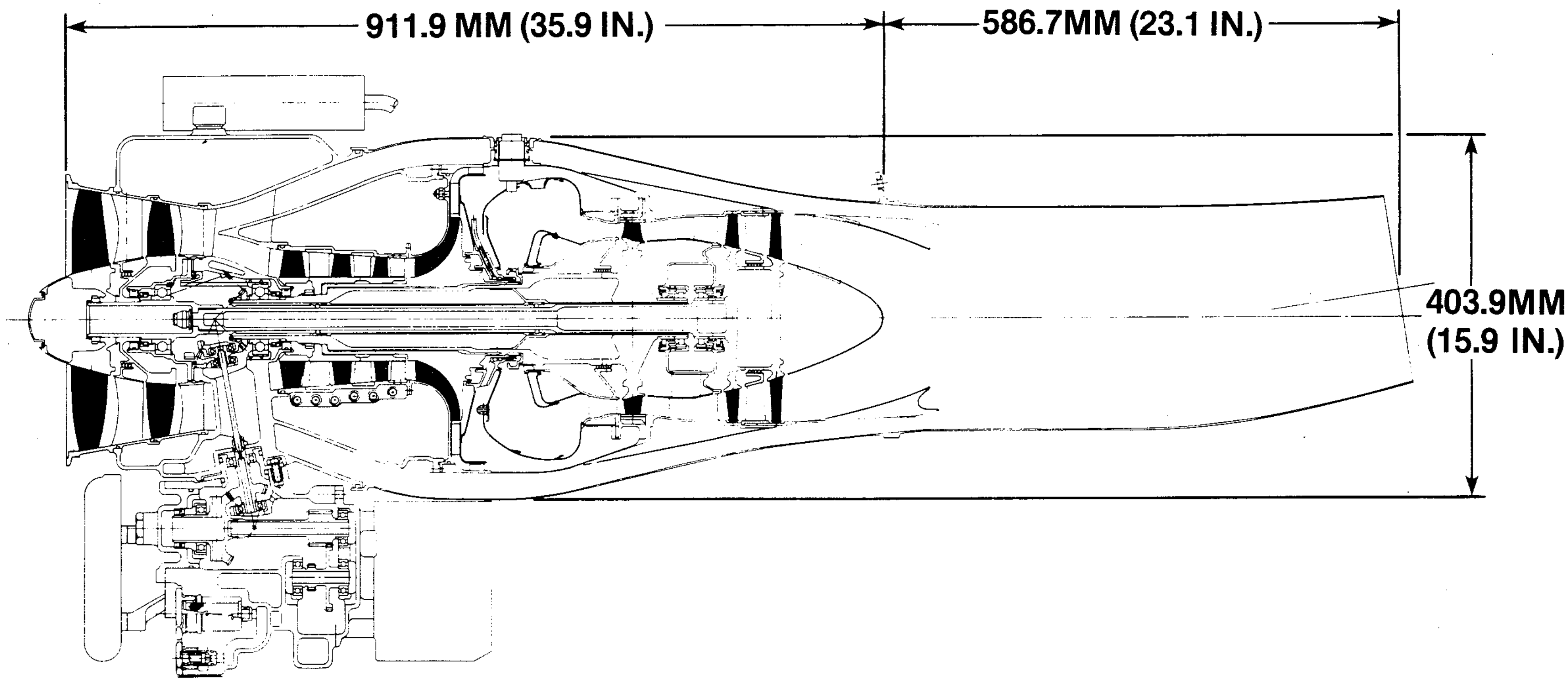


Figure 5. Cruise Missile Vehicle Drag Characteristics.



SL/0.7 PERFORMANCE (INLET REC. = 1.0)

THRUST, N (LB)	4448(1000)	BPR	1.11
SFC, KG/HR/N (LB/HR/LB)	0.102(1.005)	OPR	12.6
TOTAL AIRFLOW, KG/SEC (LB/SEC)	12.8(28.2)	TRIT,°C(°F)	1127 (2060)

THRUST/WT. (4.4 LB/LB) 43.1 N/KG

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Figure 6. Baseline (State-Of-The-Art) Engine.

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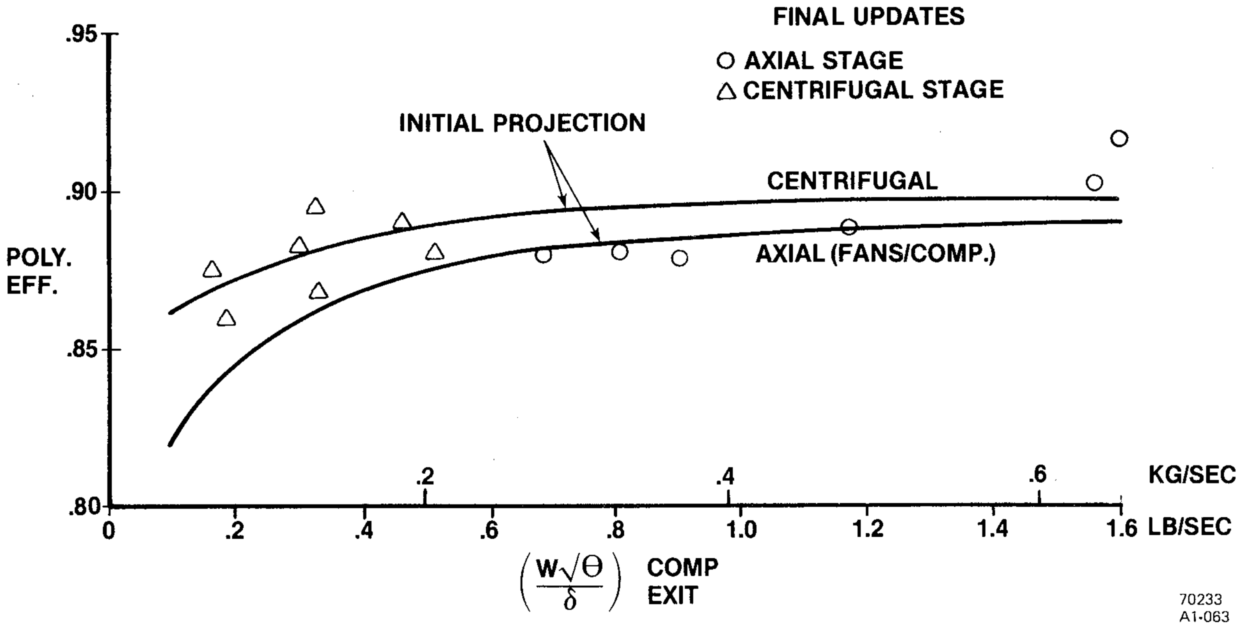


Figure 7. Preliminary Fan/Compressor Polytropic Efficiency Projections To YR 2000 (Corrected Flow Up To 0.725 kg/sec).

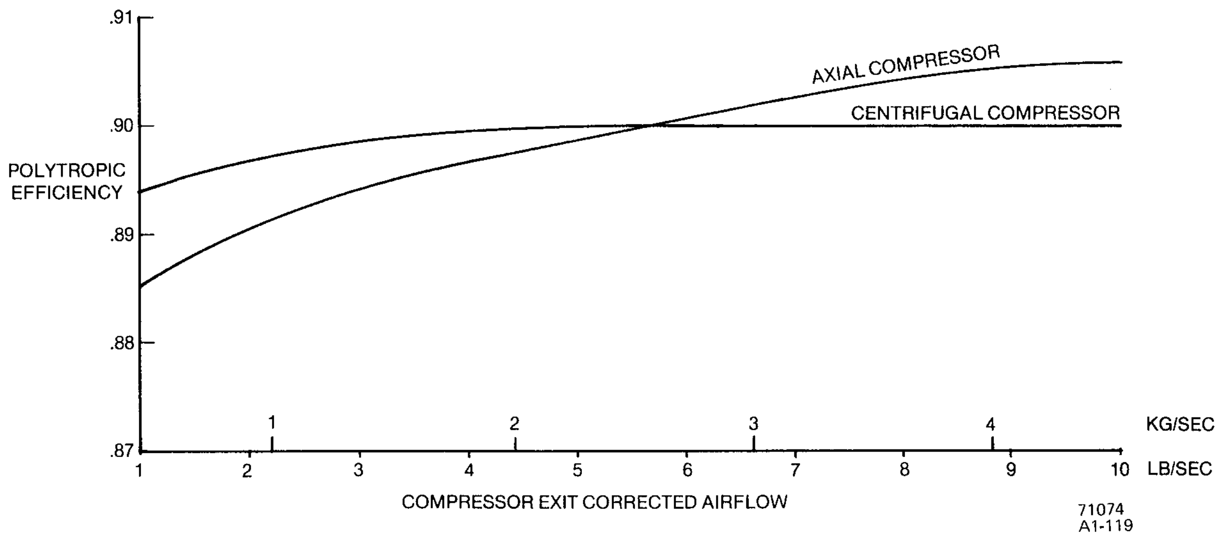


Figure 8. Preliminary Fan/Compressor Polytropic Efficiency Projections To YR 2000 (Correct Flow Up To 4.536 kg/sec).

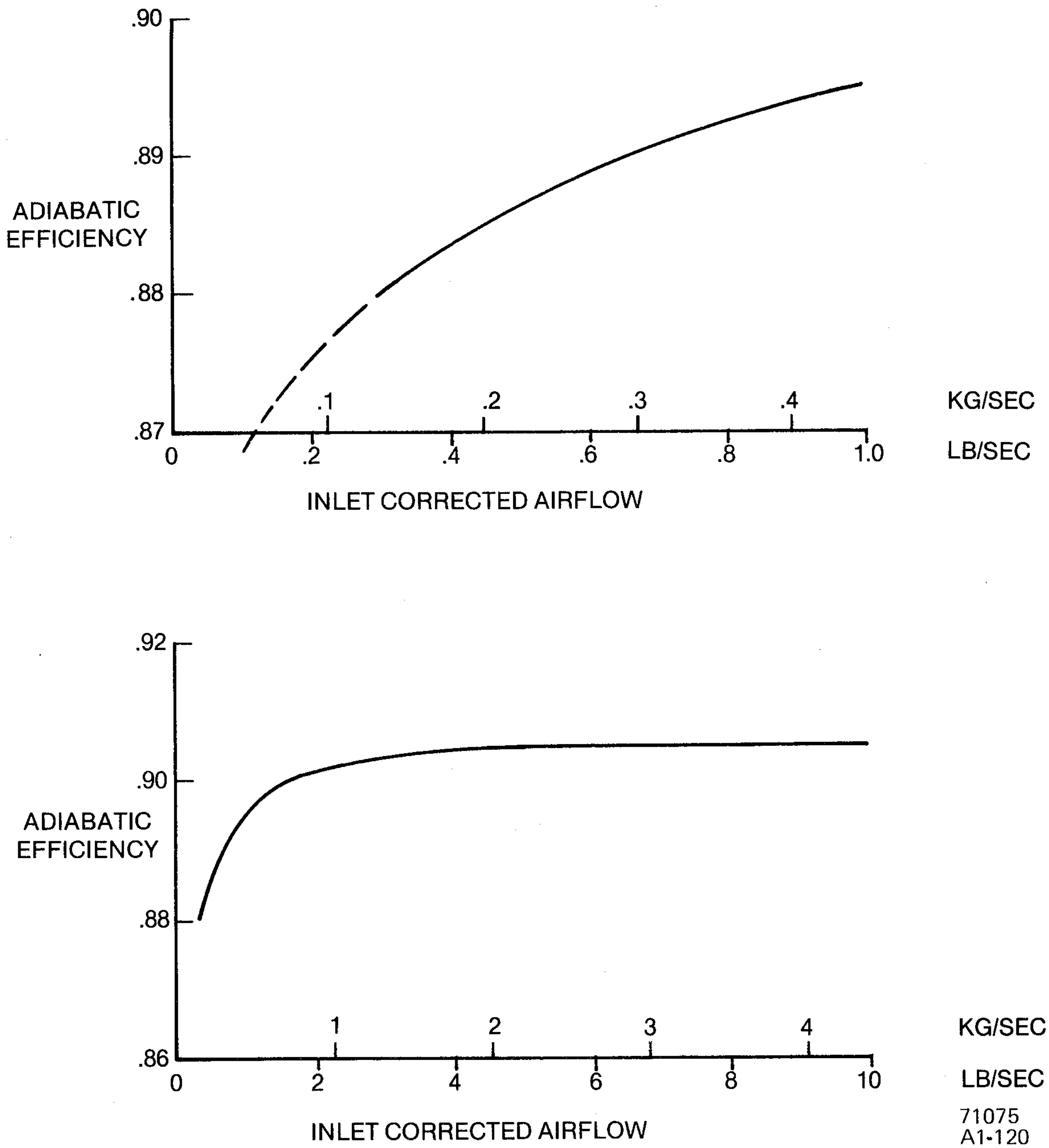


Figure 9. Preliminary Radial Turbine Efficiency Projections To YR 2000.

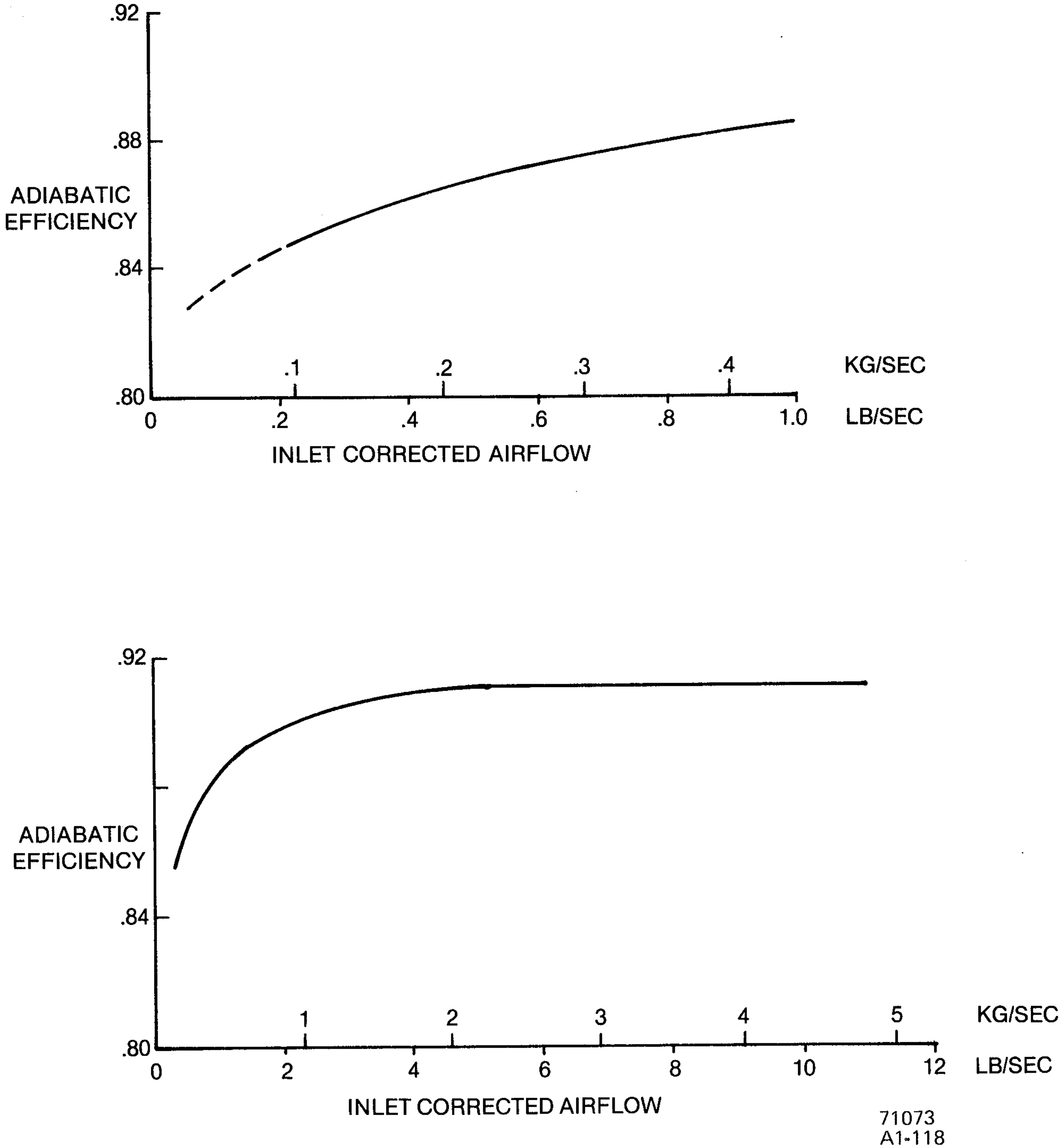


Figure 10. Preliminary Axial Turbine Efficiency Projections To YR 2000.

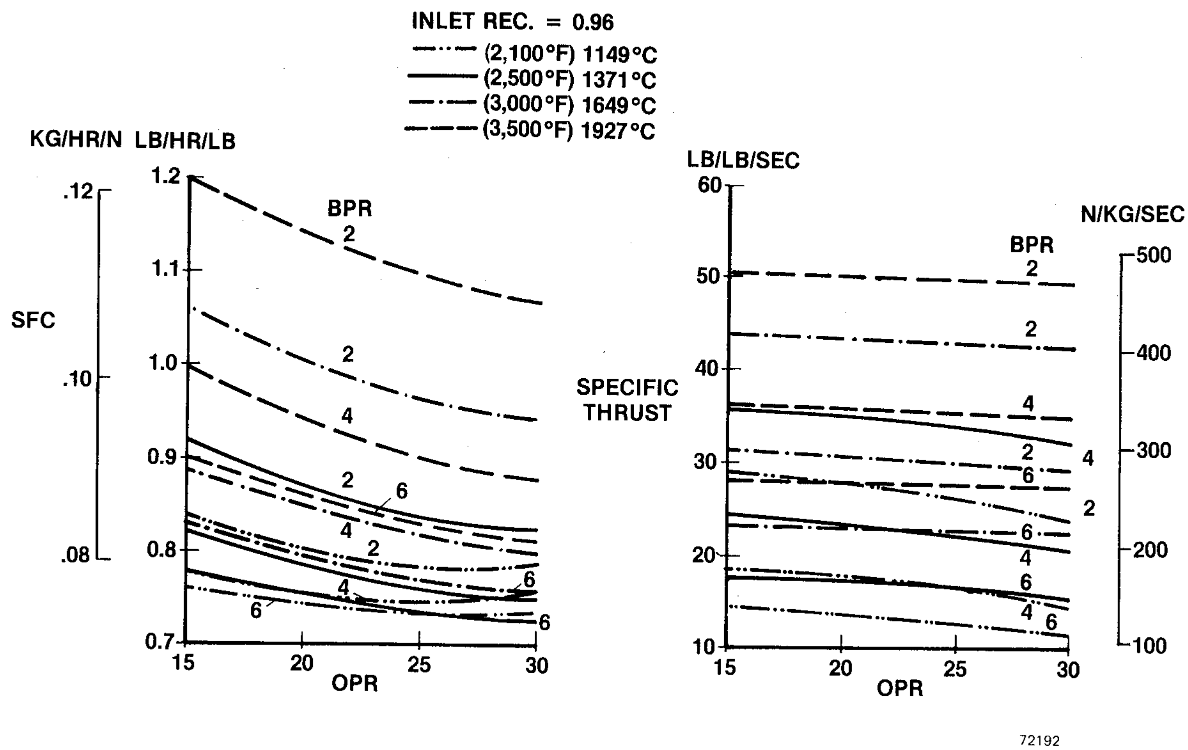


Figure 11. Two-Spool Turbofan Parametric Performance: Sea Level, Mach 0.7.

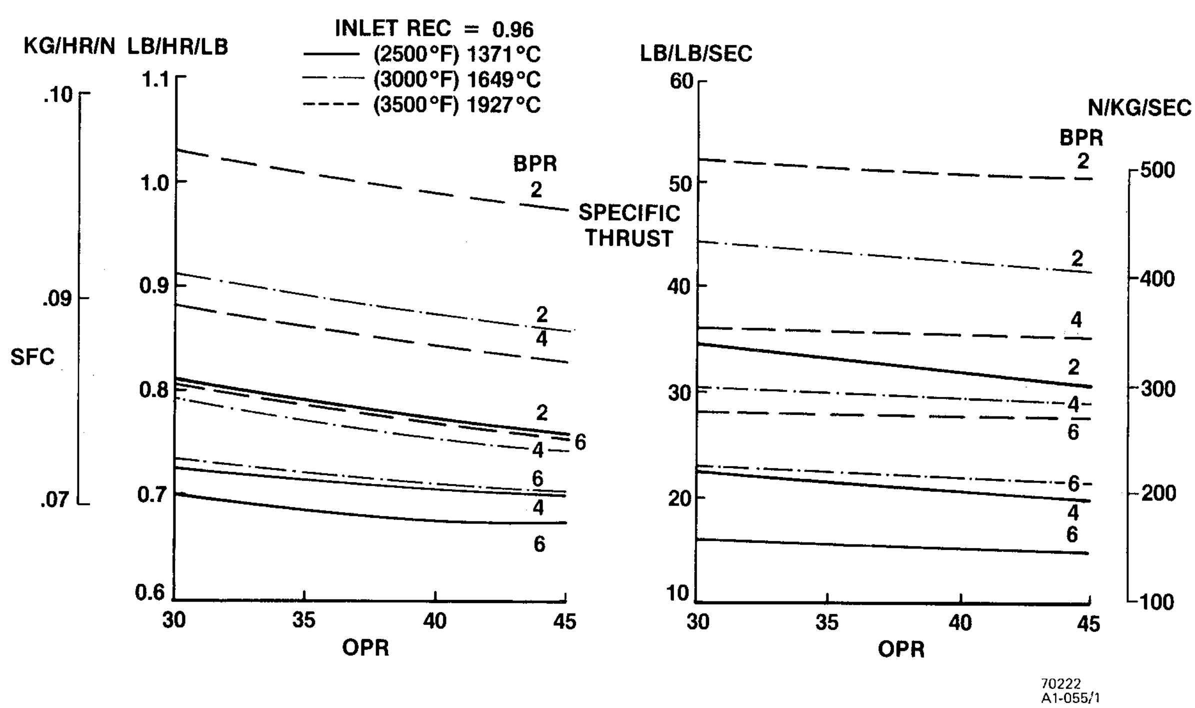


Figure 12. Three-Spool Turbofan Parametric Performance: Sea Level, Mach 0.7.

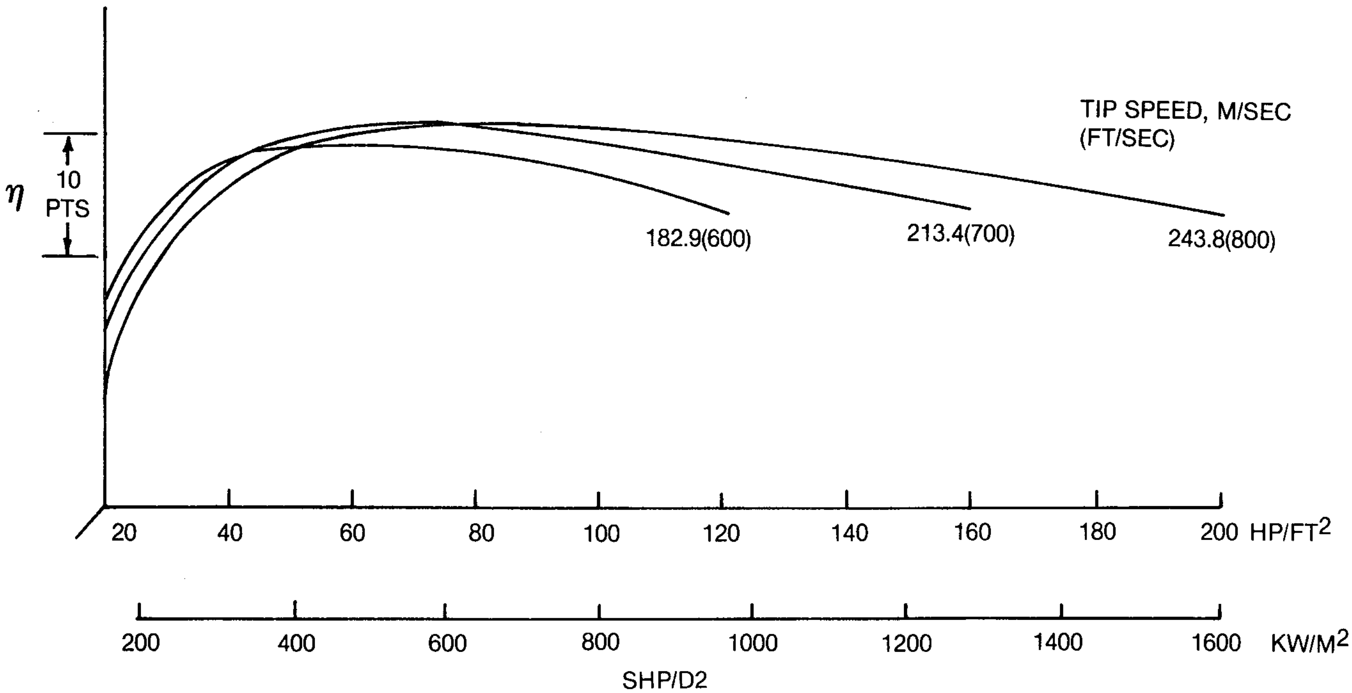


Figure 13. 4x4 Counter Rotating Propeller Performance.

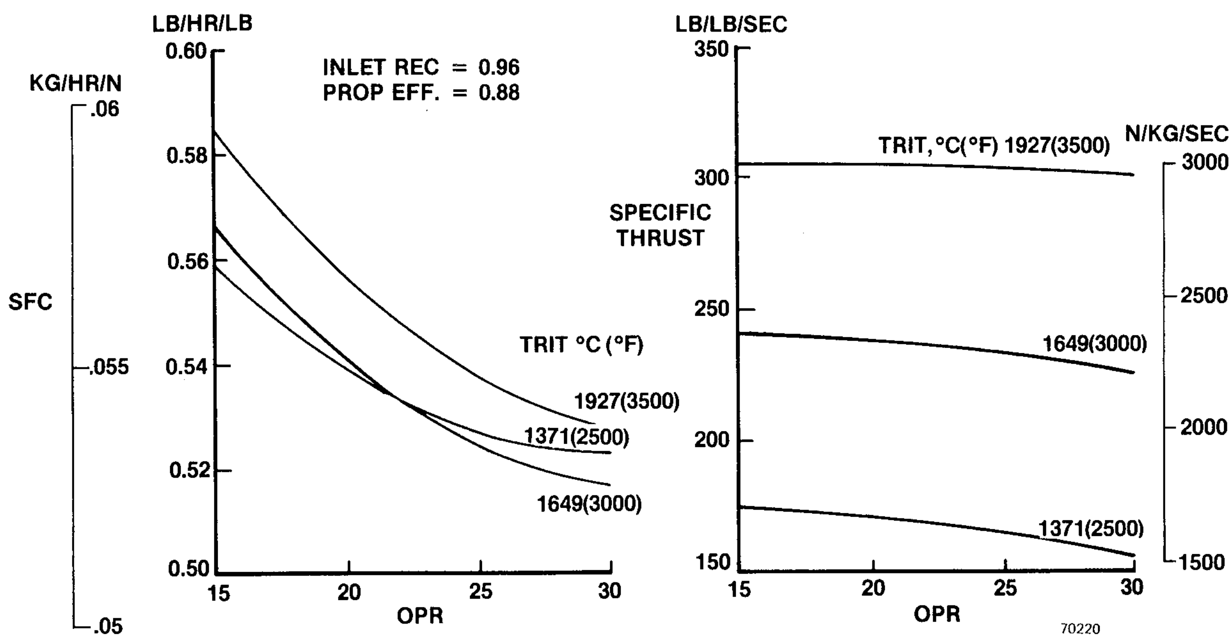


Figure 14. Two-Spool Propfan Parametric Performance: Sea Level, Mach 0.7.

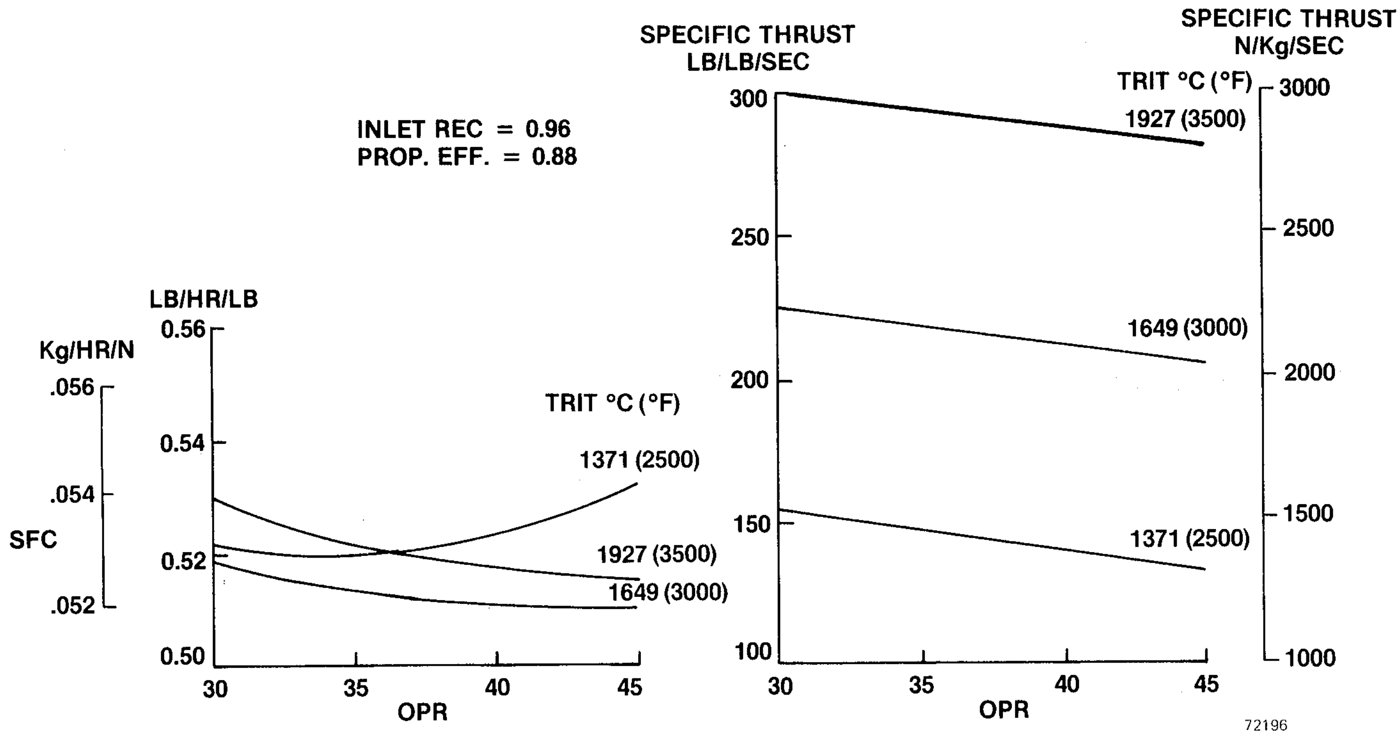


Figure 15. Three-Spool Propfan Parametric Performance: Sea Level, Mach 0.7.

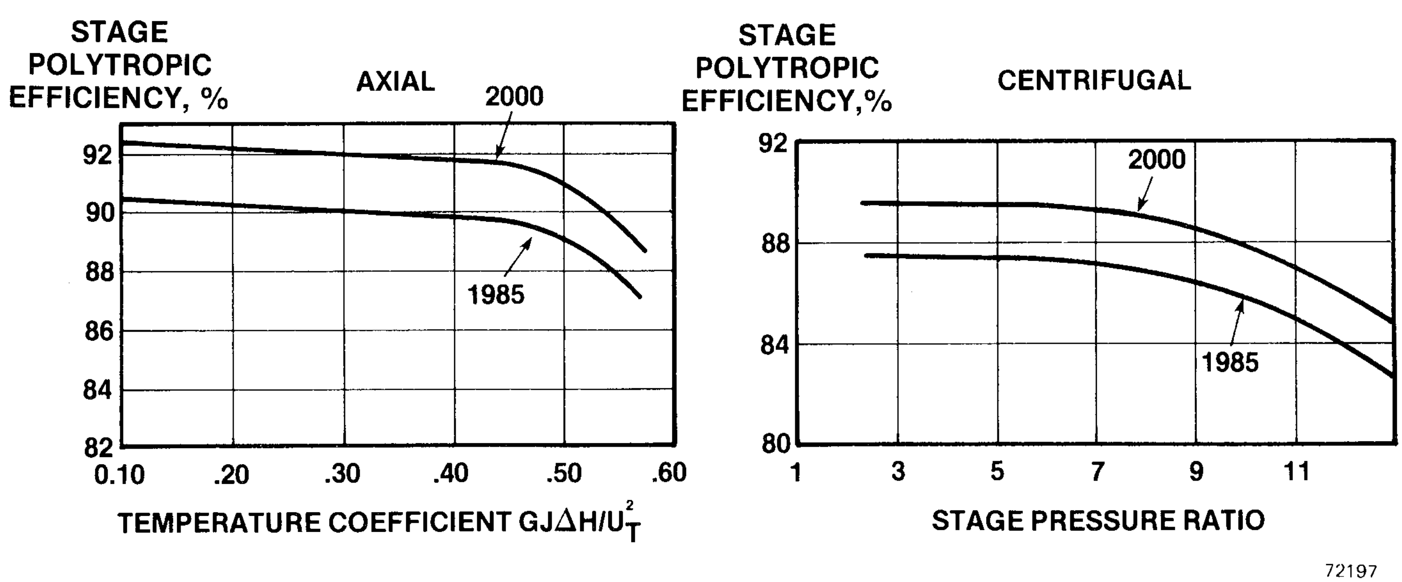


Figure 16. Compressor Polytropic Efficiency Projections to YR 2000.

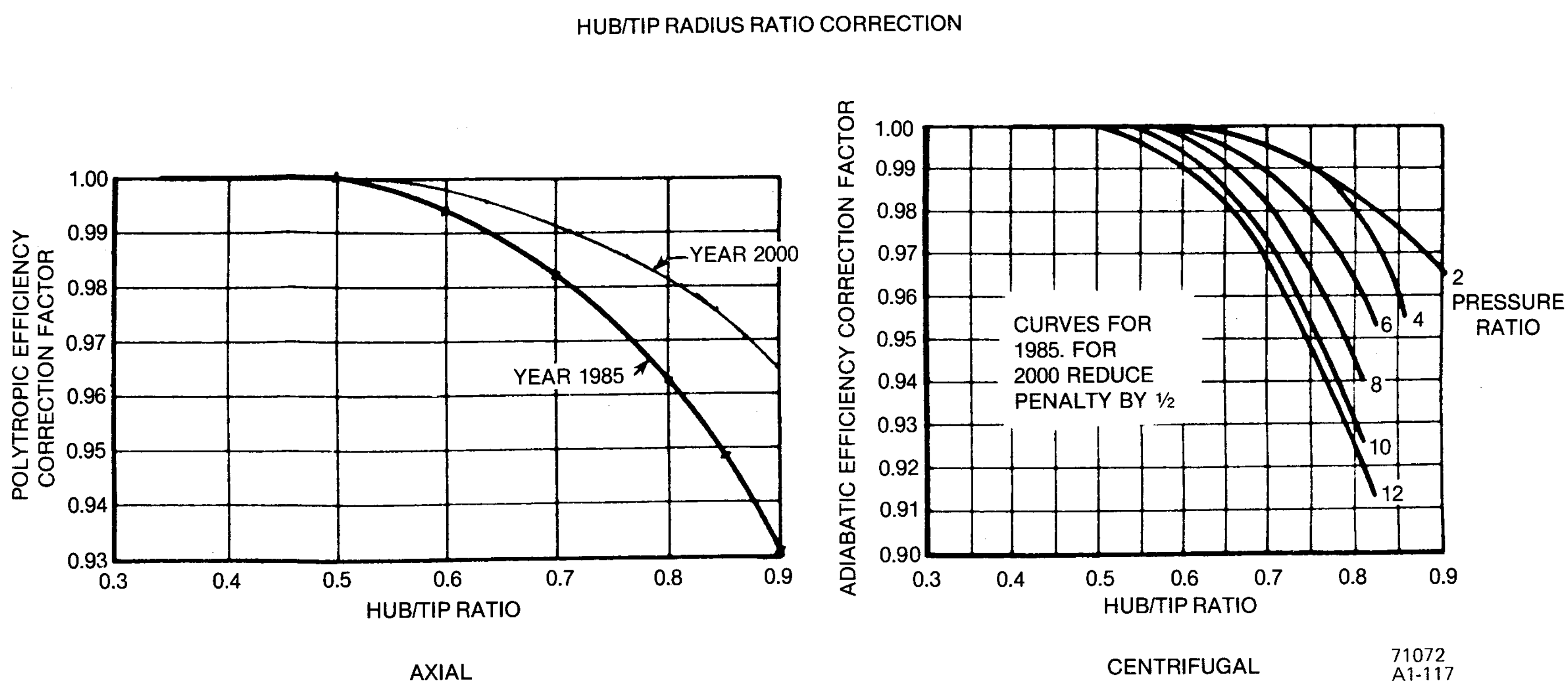


Figure 17. Compressor Hub/Tip Ratio Corrections For Efficiency.

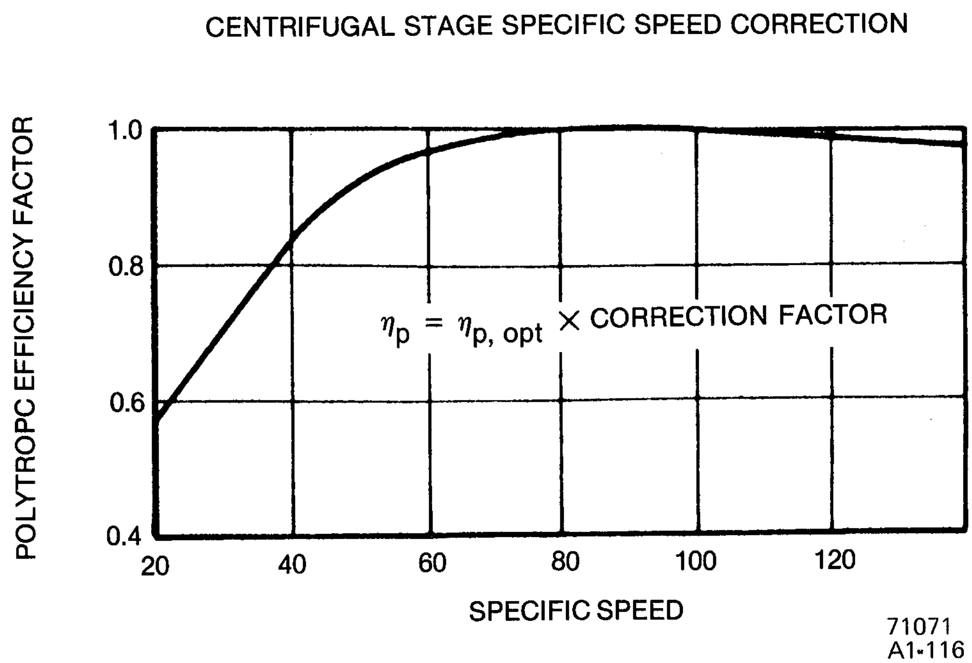


Figure 18. Centrifugal Compressor Specific Speed Corrections For Efficiency.

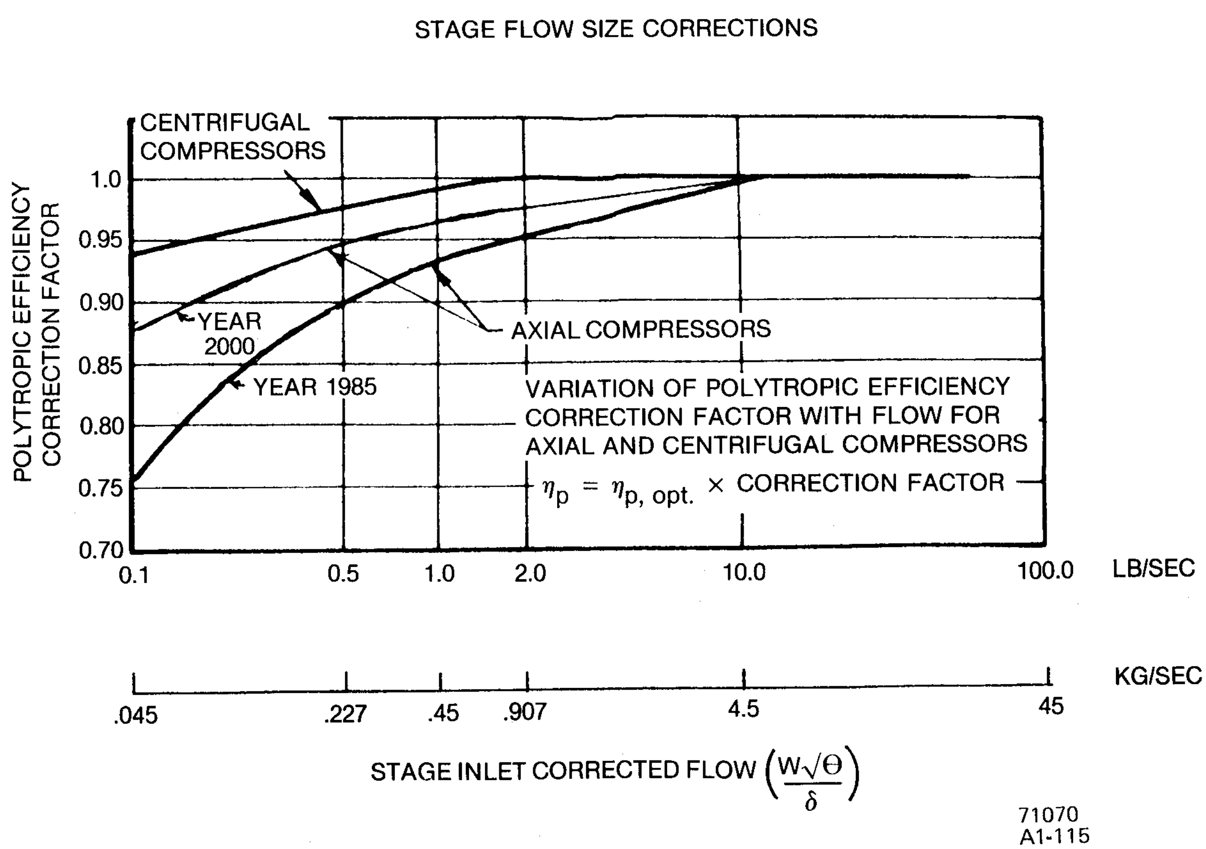


Figure 19. Compressor Flow Size Corrections For Efficiency.

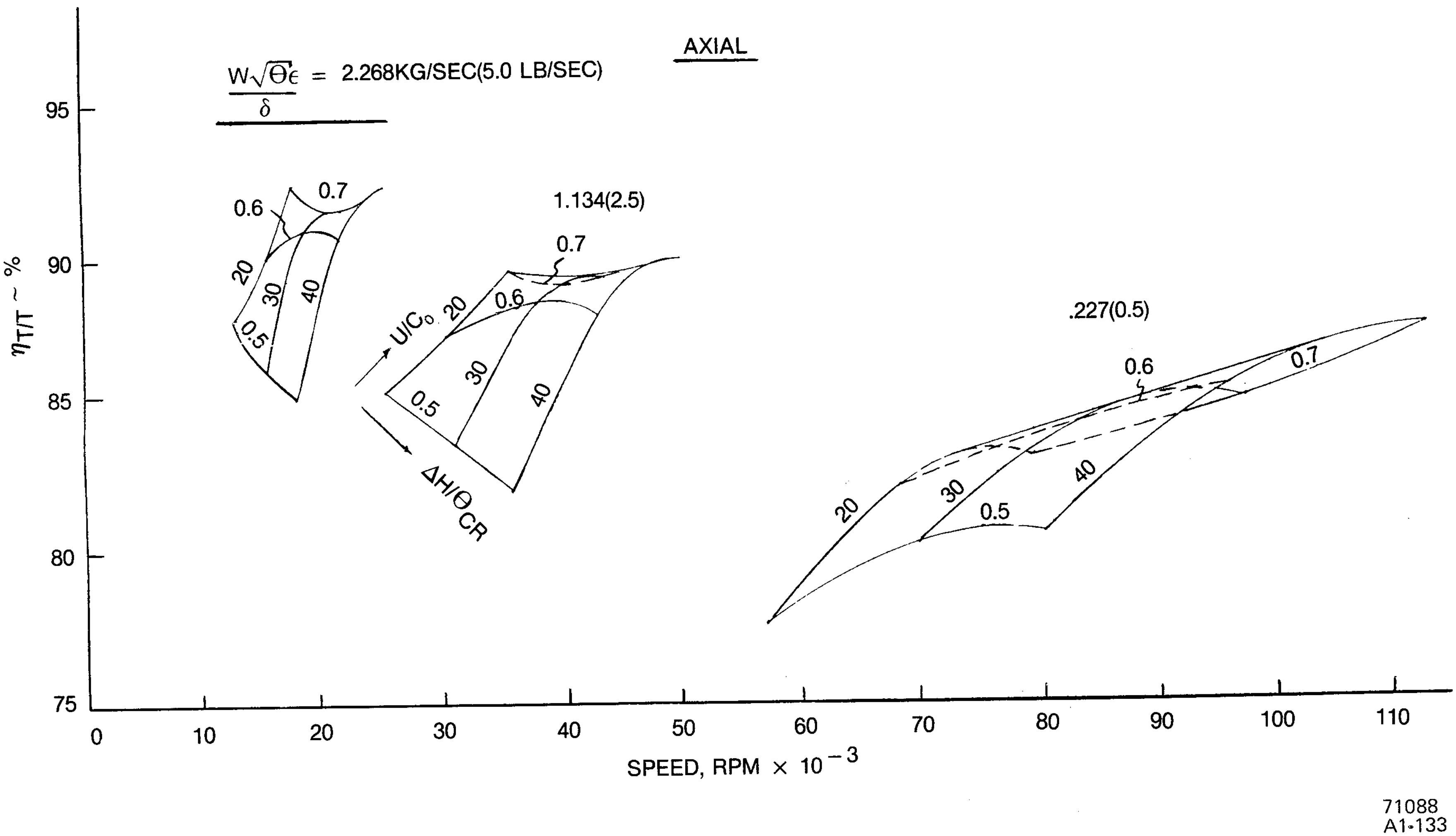


Figure 20. Axial Turbine Efficiency Projections To YR 2000.

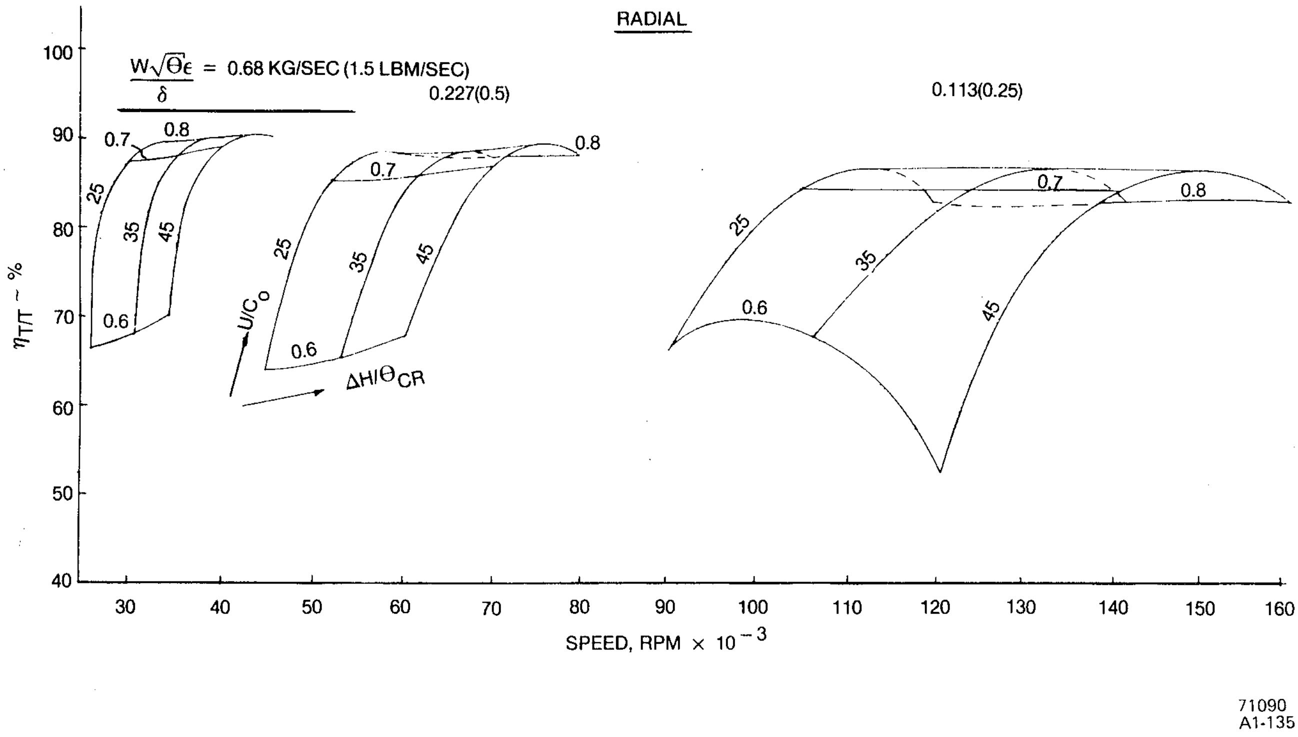


Figure 21. Radial Turbine Efficiency Projections To YR 2000.

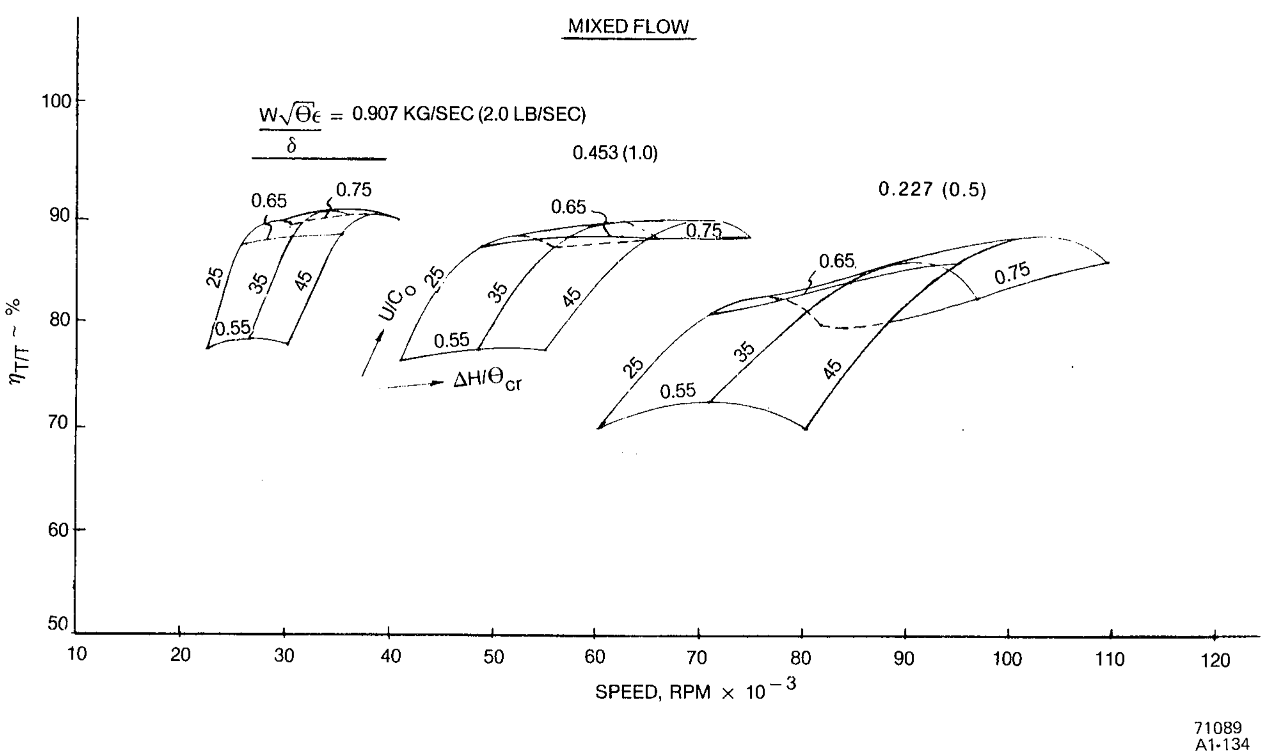
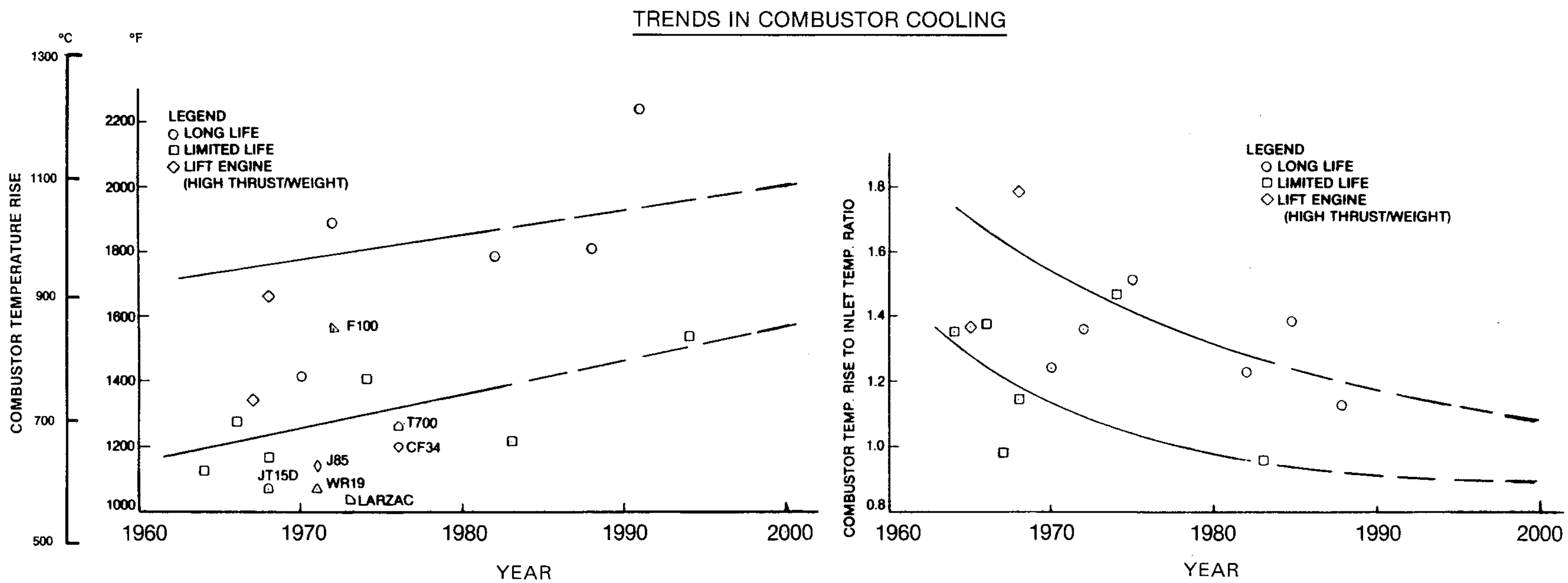


Figure 22. Mixed Flow Turbine Efficiency Projections To YR 2000.



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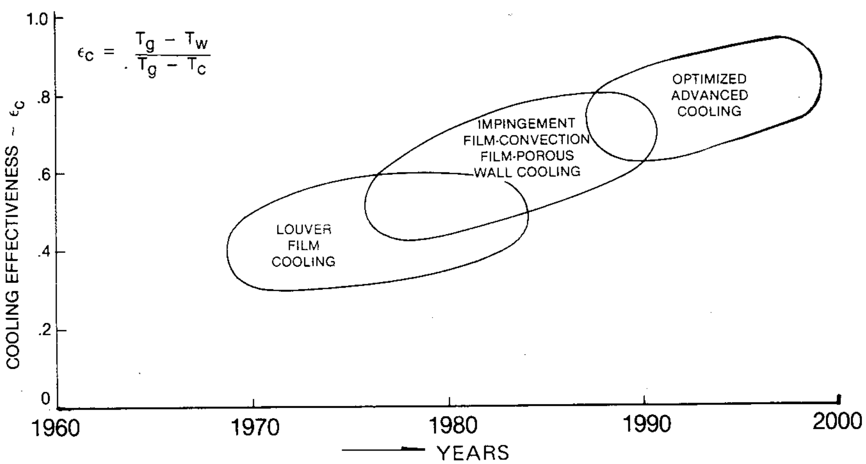
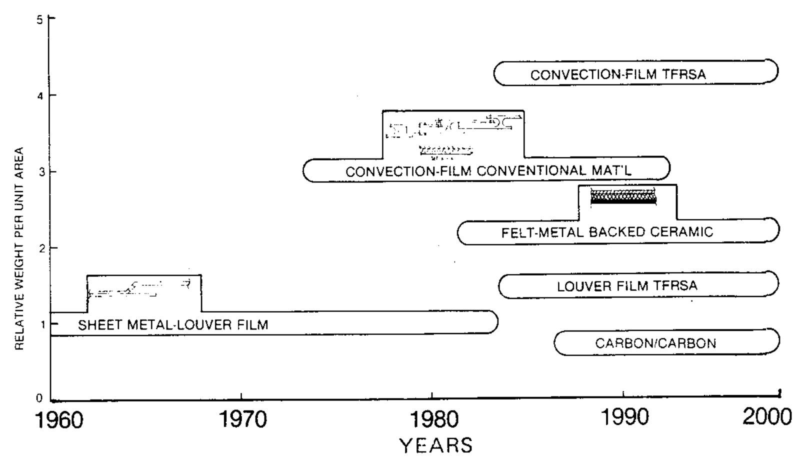
A1-156

Figure 23. Combustor Temperature Trends.

TRENDS IN COMBUSTOR TEMPERATURES





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A1-155

Figure 24. Combustor Cooling Effectiveness Trends.

Figure 25. Combustor Weight Trends.

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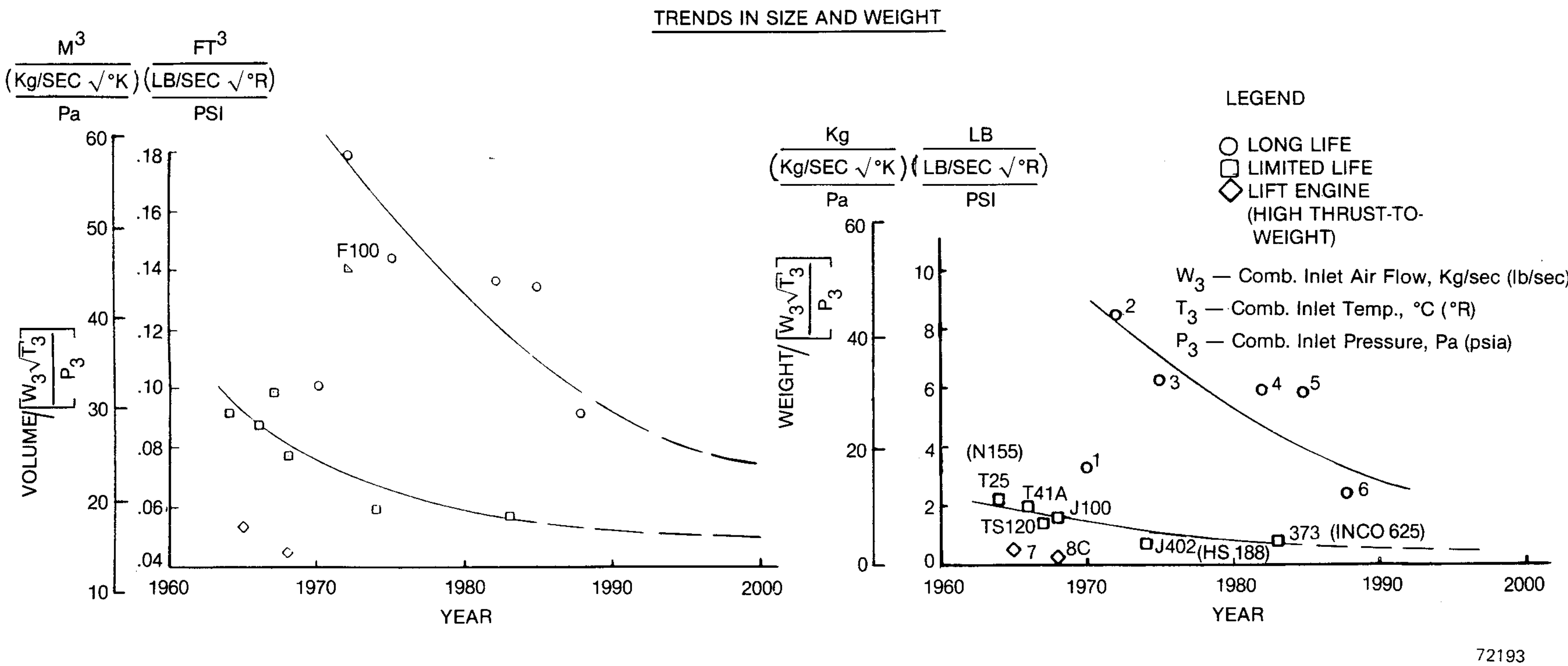


Figure 26. Combustor Size and Weight Trends.

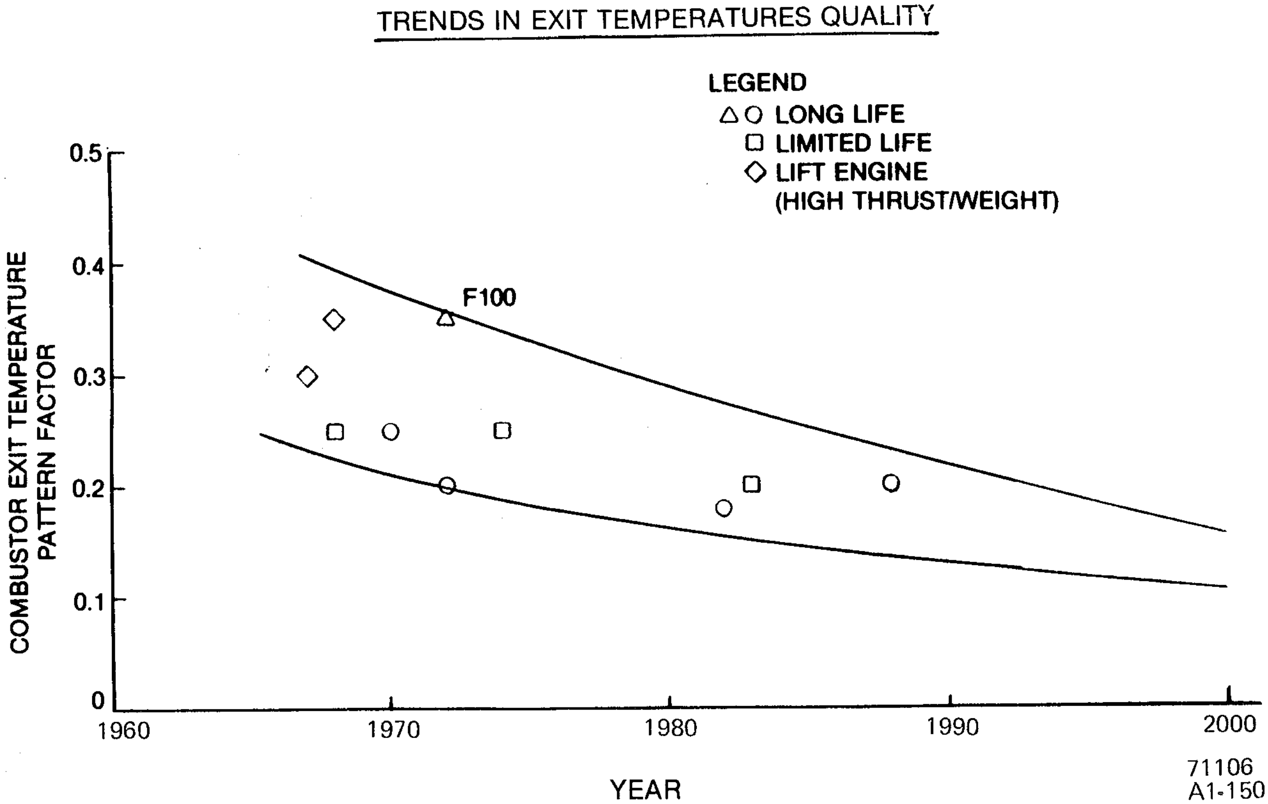


Figure 27. Combustor Exit Temperature Pattern Factor Trends.

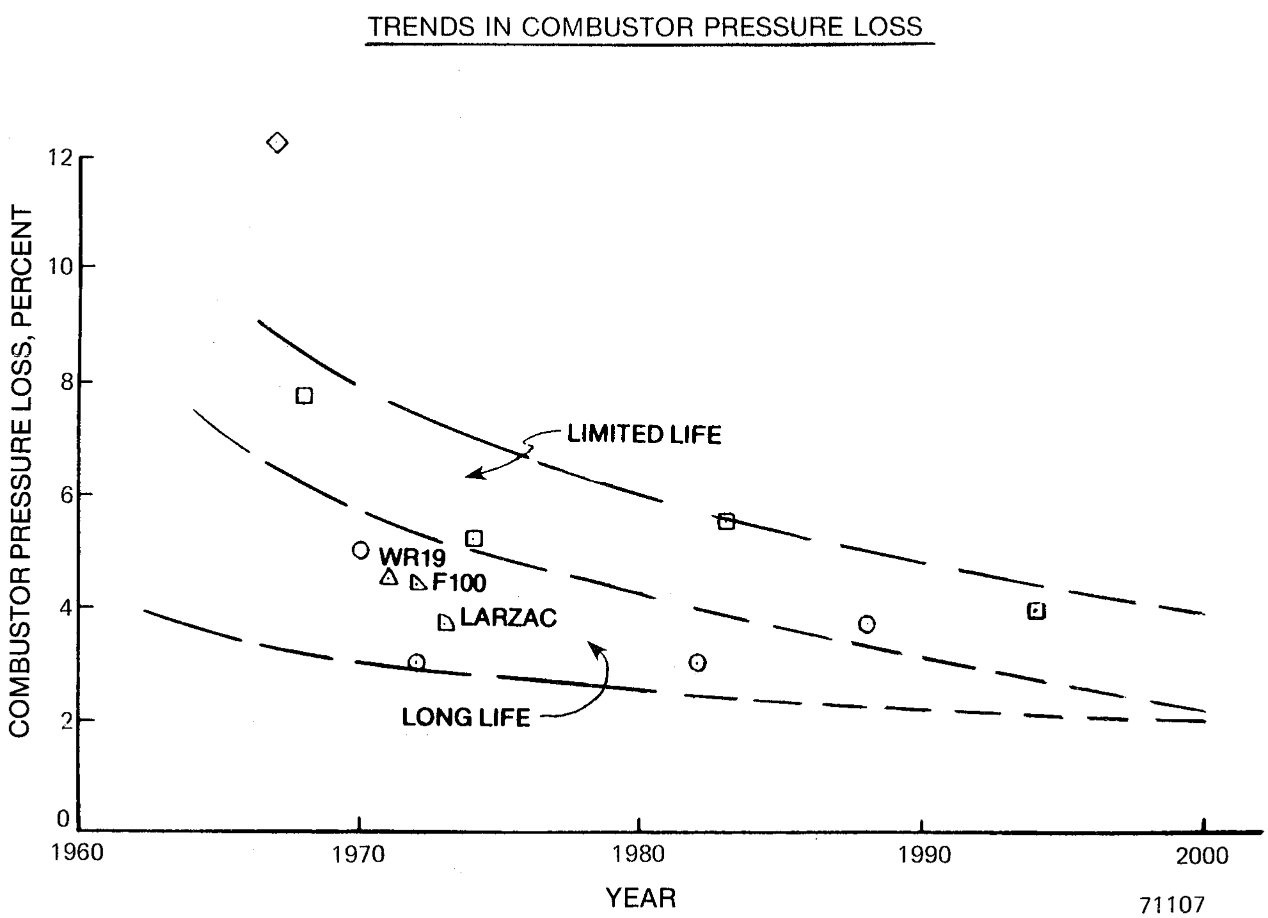
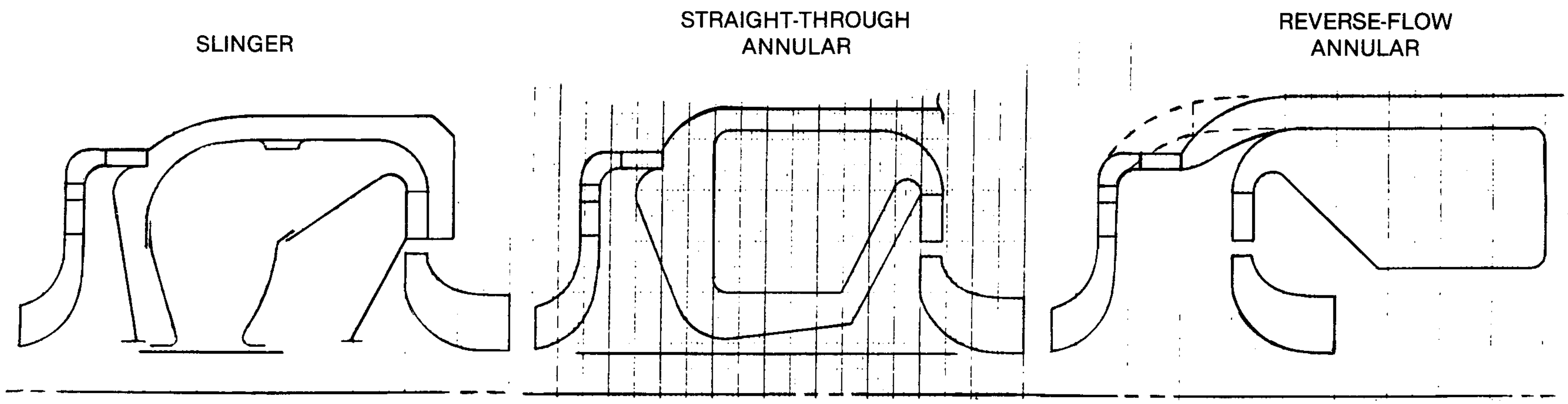


Figure 28. Combustor Pressure Loss Trends.



	SLINGER	STRAIGHT-THROUGH ANNULAR	REVERSE-FLOW ANNULAR
	τ — 4.29 MILLISEC	τ — 4.32 MILLISEC	τ — 5.67 MILLISEC
I, J/SEC M^3 Pa $\times 10^6$	100.1	99.2	75.7
(MBTU/HR — FT^3 — ATM)	(9.8)	(9.71)	(7.41)
SPEC. VOL, $\left(\dfrac{M^3}{\frac{Kg/SEC.\ \sqrt{°K}}{Pa}}\right)$	21.9	22.1	29.0
$\left(\dfrac{FT^3}{\frac{LB/SEC\ \sqrt{°R}}{PSI}}\right)$	(0.683)	(.0689)	(.0904)

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Figure 29. Combustor Configurations.

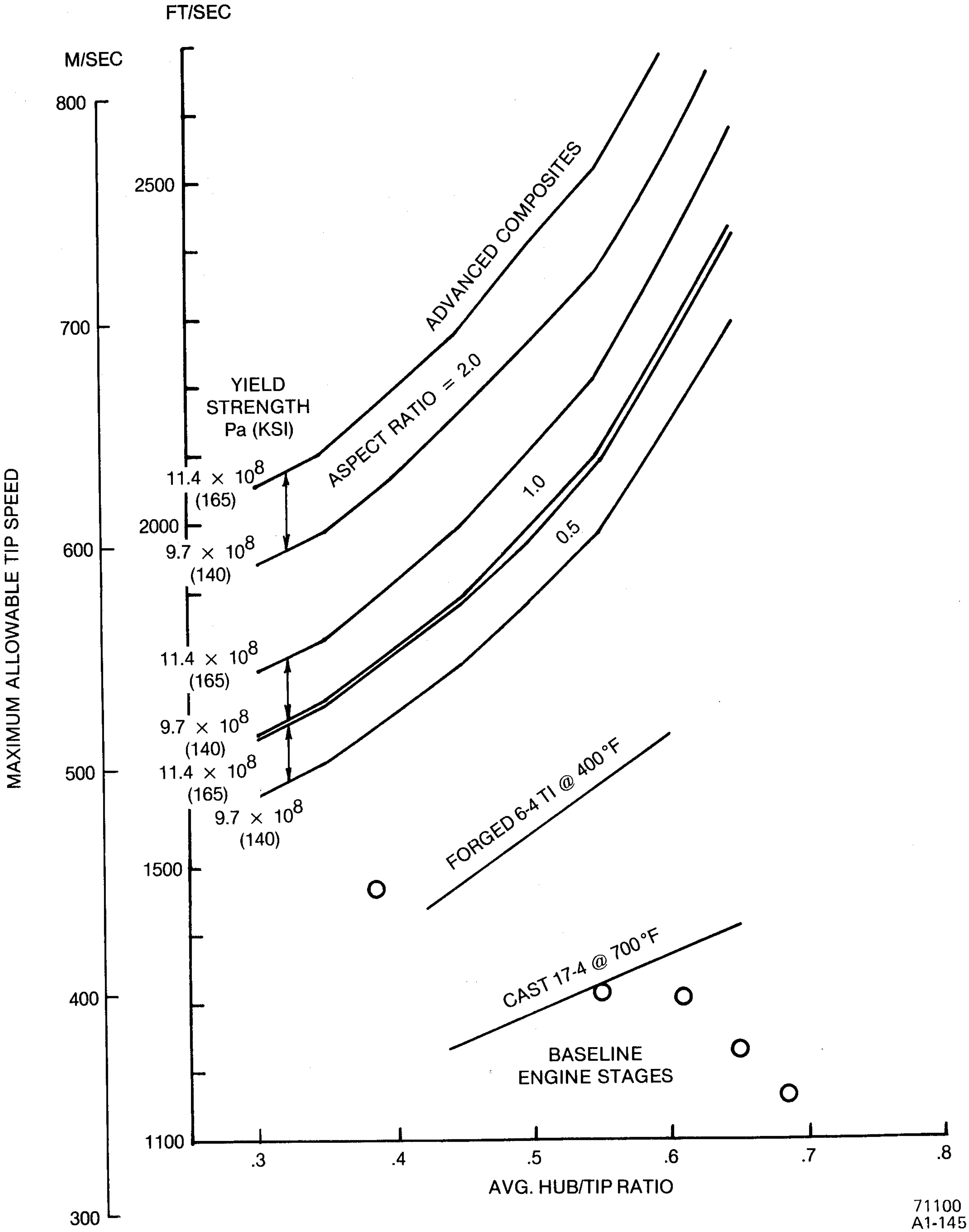

Figure 30. Axial And Mixed Flow Fan/Compressor Structural Guidelines: Maximum Allowable Tip Speed Projections.

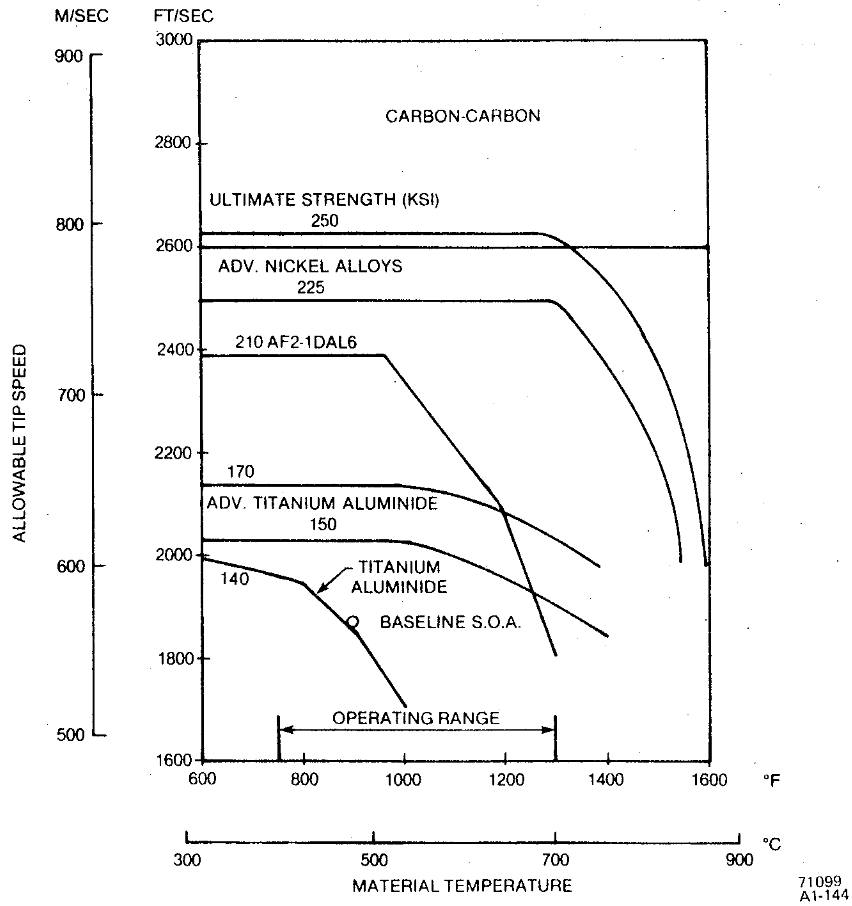

Figure 31. Centrifugal And Mixed Flow Compressor Structural Guidelines: Maximum Allowable Tip Speed Projections.

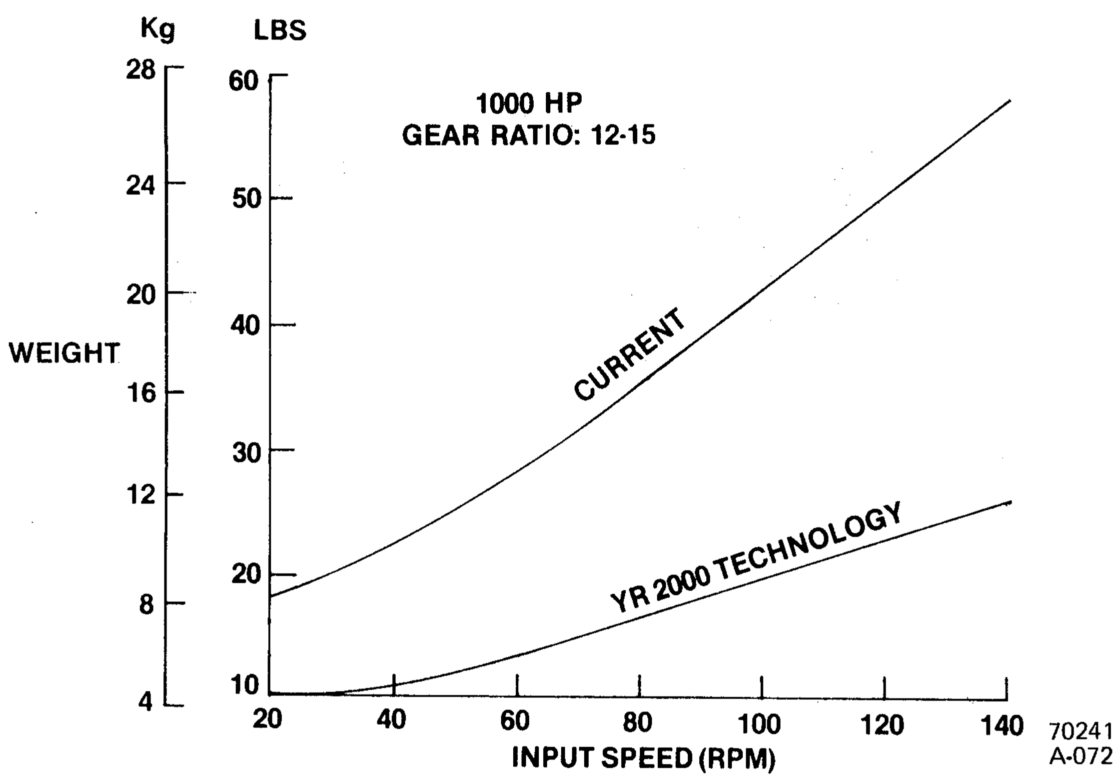

Figure 32. Propfan Gearbox Weight Projections: Gearbox Weight VS Input Speed.

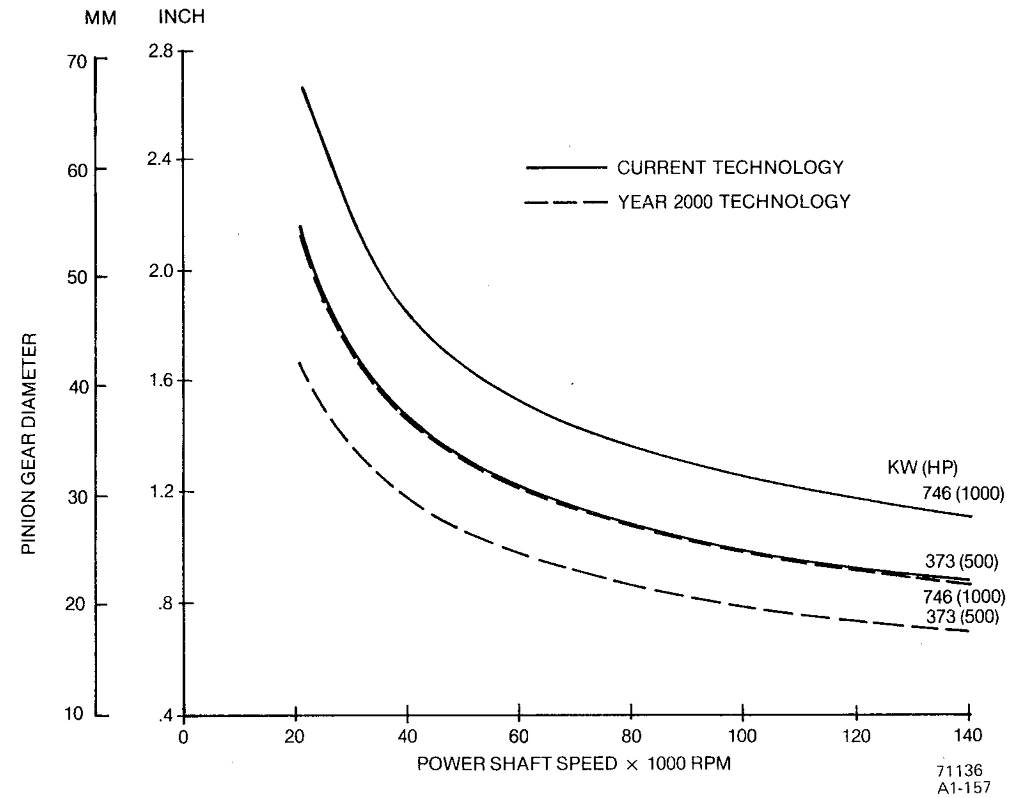

Figure 33. Propfan Gearbox Shaft Diameter Projections: Shaft Diameter VS Power Shaft Speed.

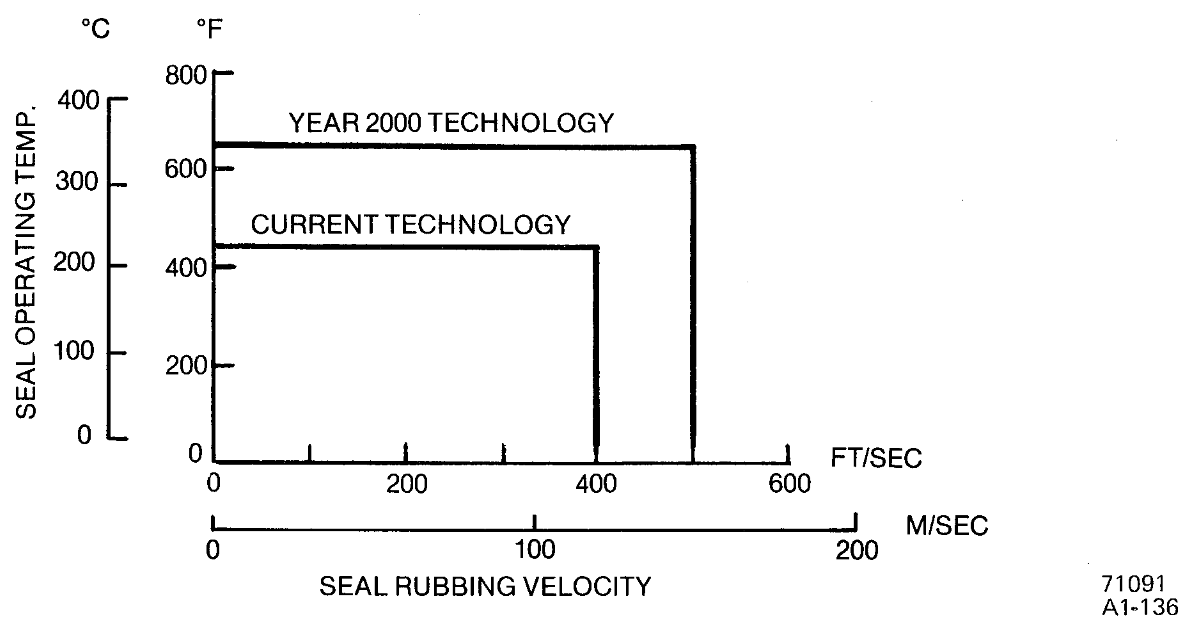

Figure 34. Contacting Seals Guidelines: Maximum Allowable Temperature VS Seal Rubbing Velocity.

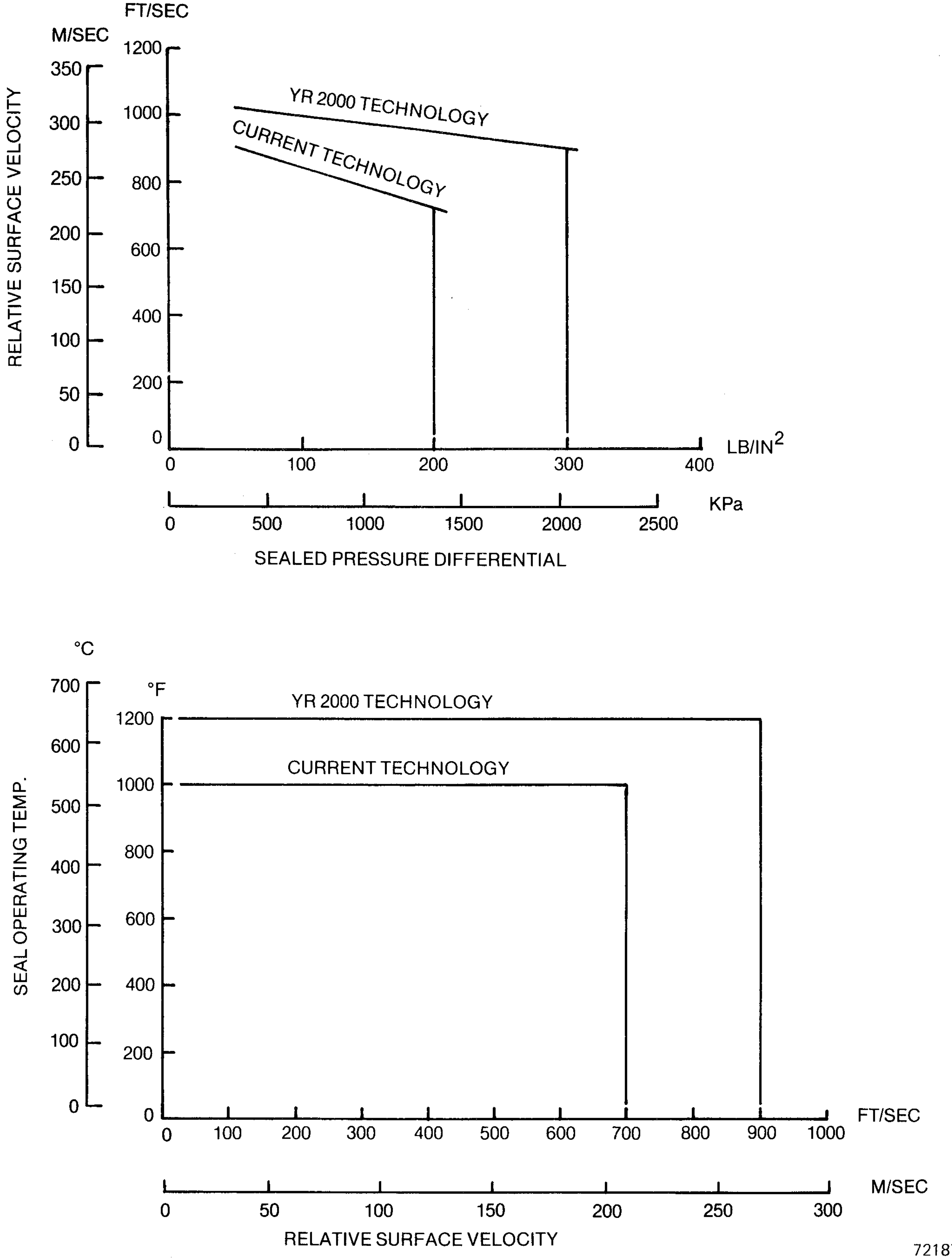

Figure 35. Non-Contacting Seals Guidelines: Maximum Allowable Temperature VS Relative Surface Velocity and Relative Surface Velocity VS Seal Pressure Differential.

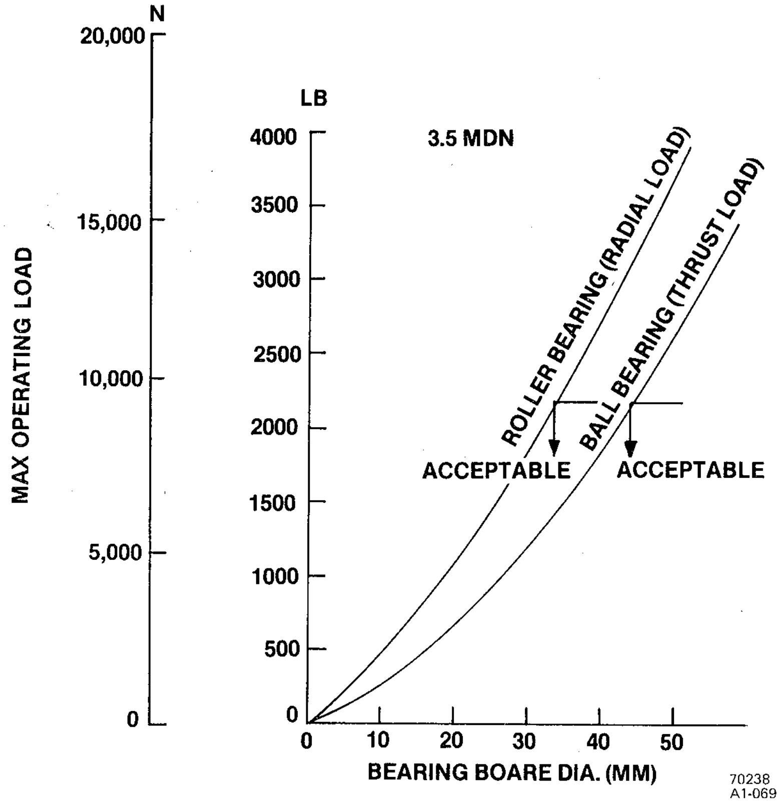

Figure 36. Bearing Design Guidelines: Maximum Operating Load VS Bearing Bore Diameter.

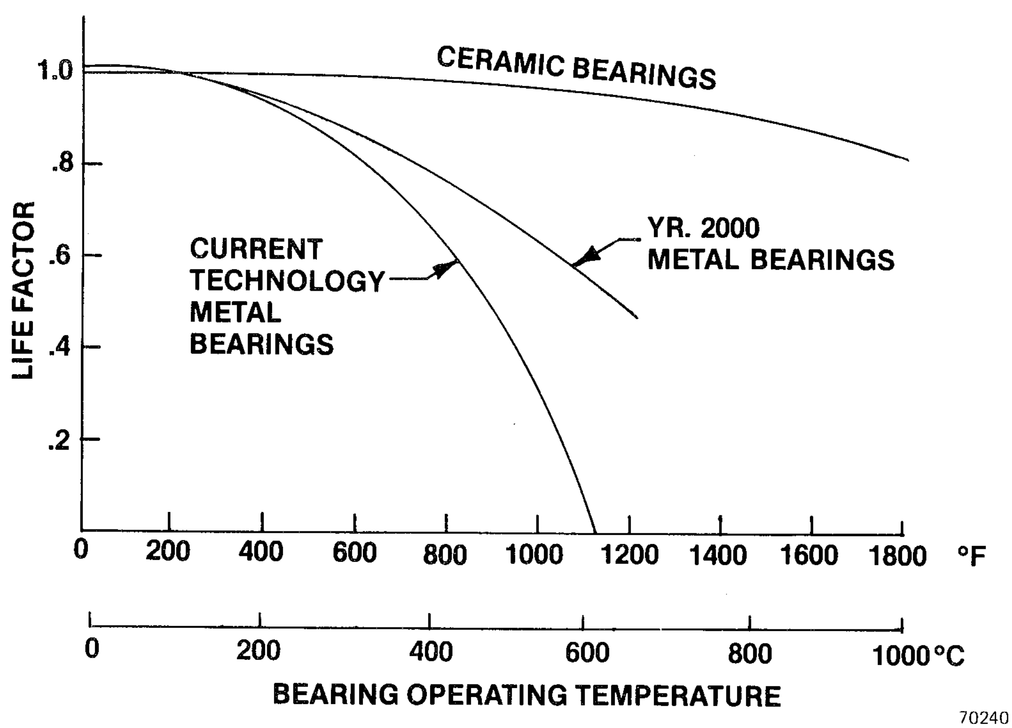

Figure 37. Bearing Design Guidelines: Life Factor VS Bearing Operative Temperature.

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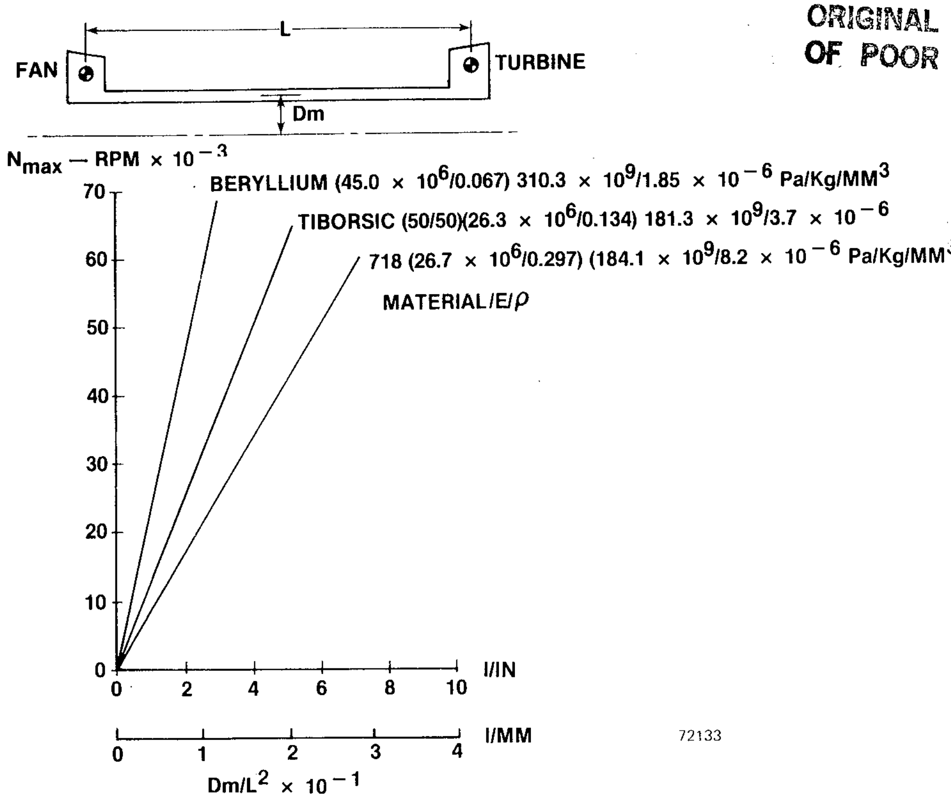

Figure 38. Low Pressure Spool Shaft Critical Speed Guidelines: Maximum Speed VS Shaft Frequency Parameter (D_m/L^2).

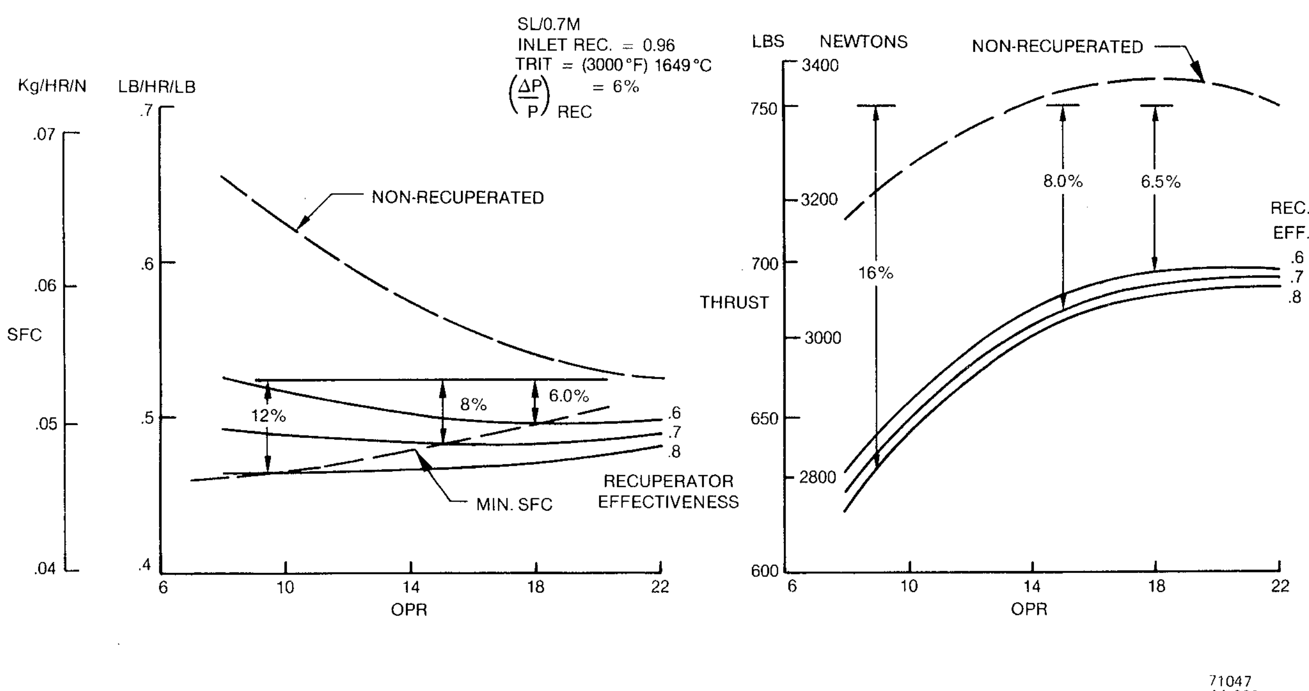

Figure 39. Two-Spool Propfan Recuperated Engine Parametric Performance: Sea Level, Mach 0.7, 6% Recuperator Pressure Loss.

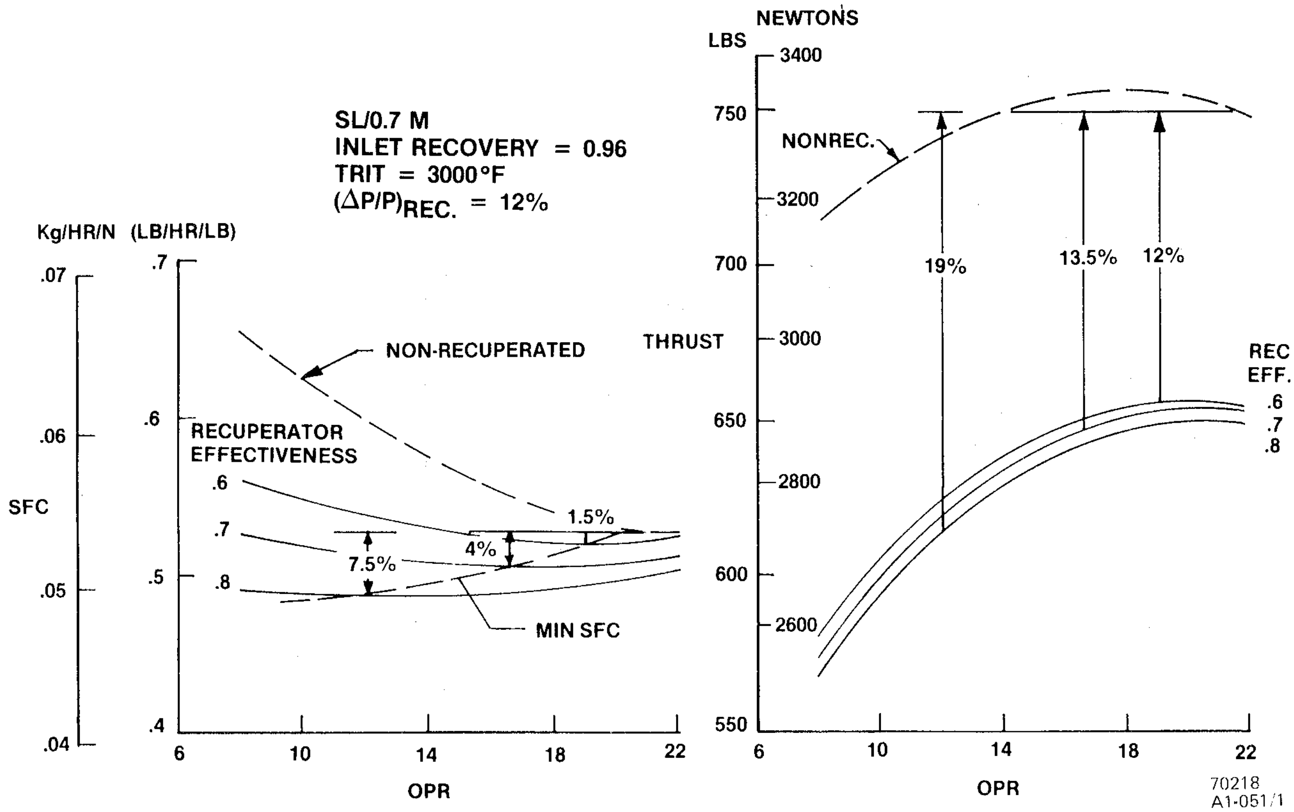

Figure 40. Two-Spool Propfan Recuperated Engine Parametric Performance: Sea Level, Mach 0.7, 12% Recuperator Pressure Loss.

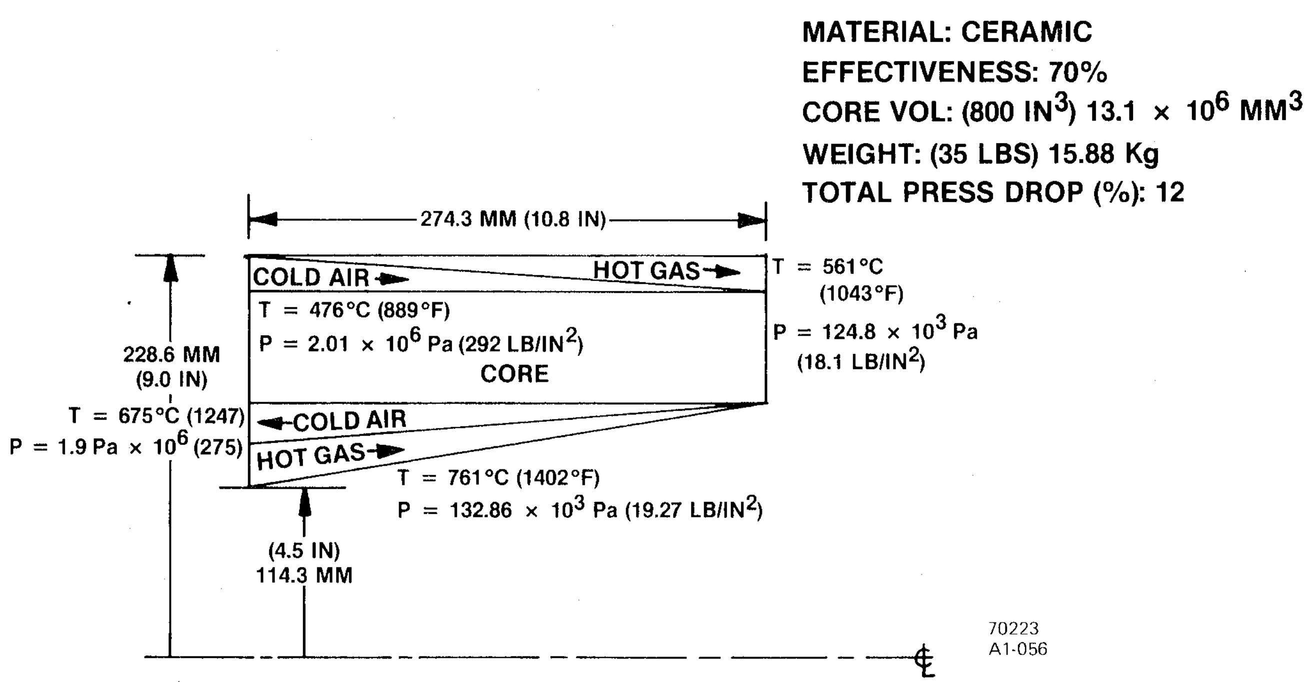

Figure 41. Recuperator Design Sketch

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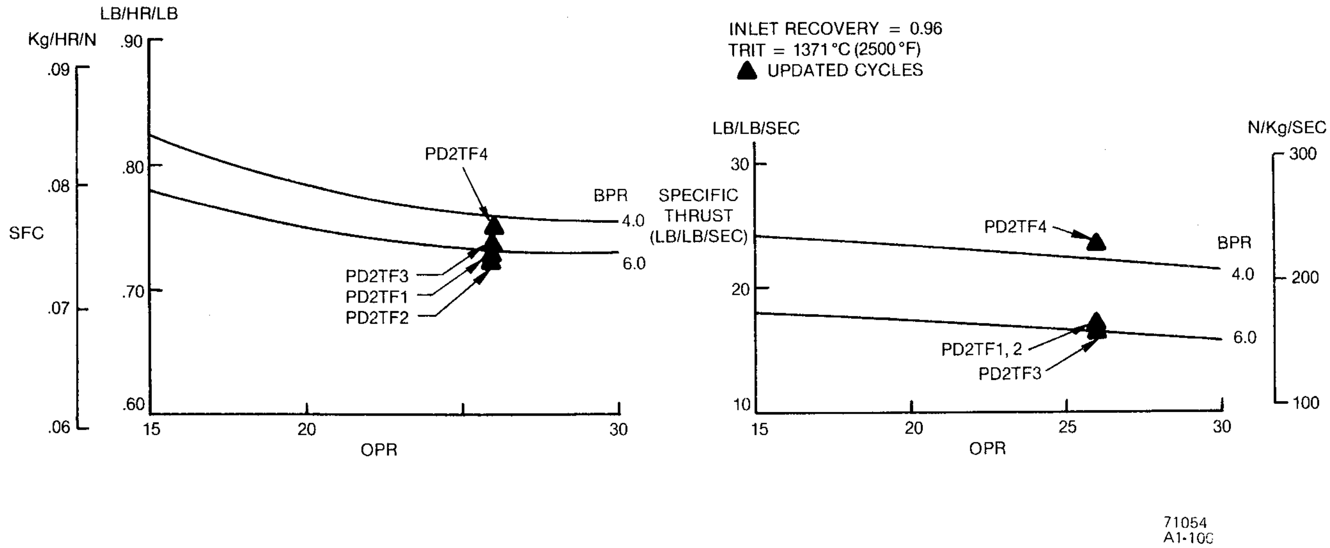

Figure 42. Two-Spool Turbofan Parametric Performance With Updated Cycles: Sea Level, Mach 0.7, 1371°C (2500°F) TRIT.

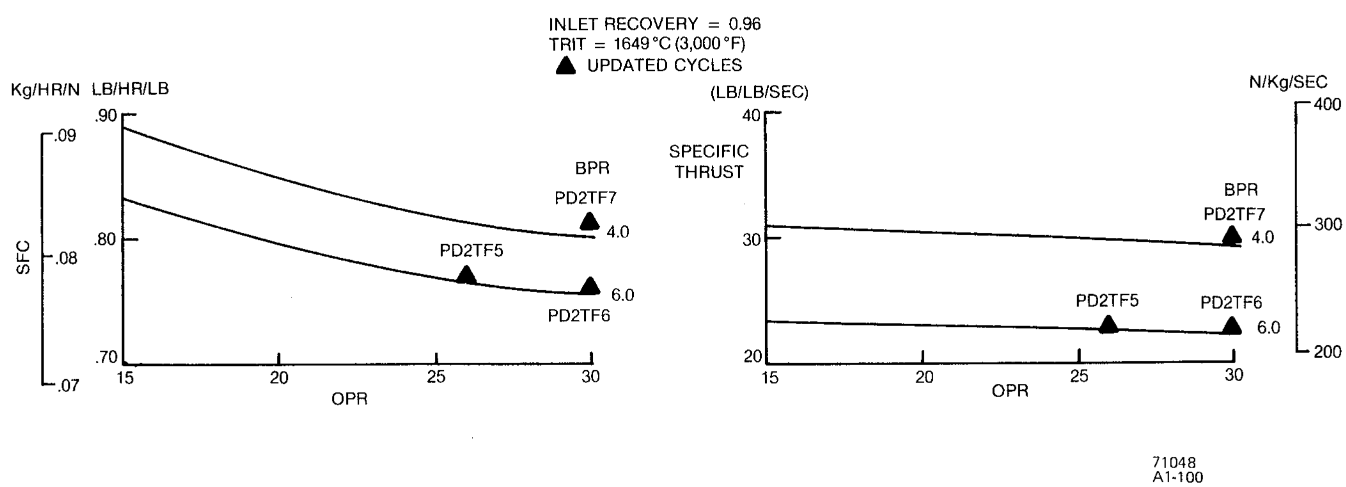

Figure 43. Two-Spool Turbofan Parametric Performance With Updated Cycles: Sea Level, Mach 0.7, 1649°C (3000°F) TRIT.

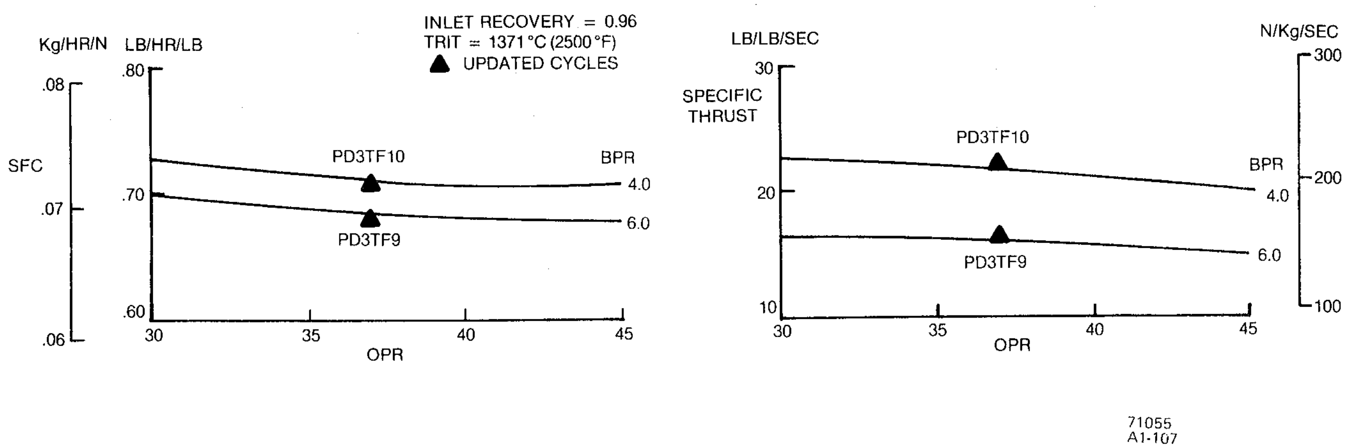

Figure 44. Three-Spool Turbofan Parametric Performance With Updated Cycles: Sea Level, Mach 0.7, 1371°C (2500°F) TRIT.

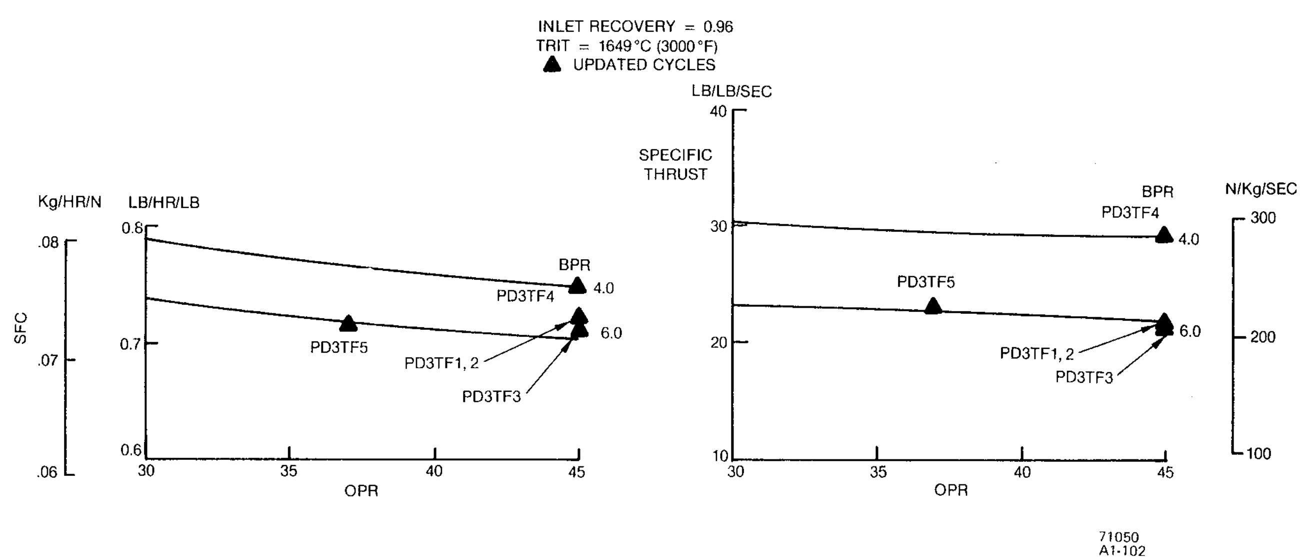

Figure 45. Three-Spool Turbofan Parametric Performance With Updated Cycles: Sea Level, Mach 0.7, 1649°C (3000°F) TRIT.

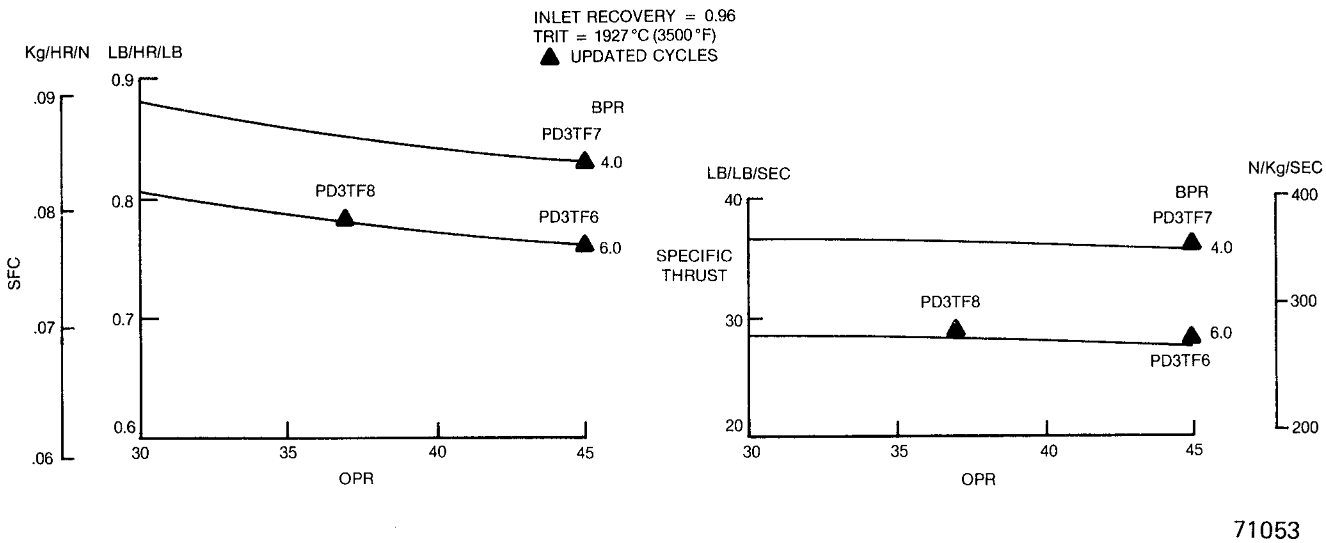

Figure 46. Three-Spool Turbofan Parametric Performance With Updated Cycles: Sea Level, Mach 0.7, 1927°C (3500°F) TRIT.

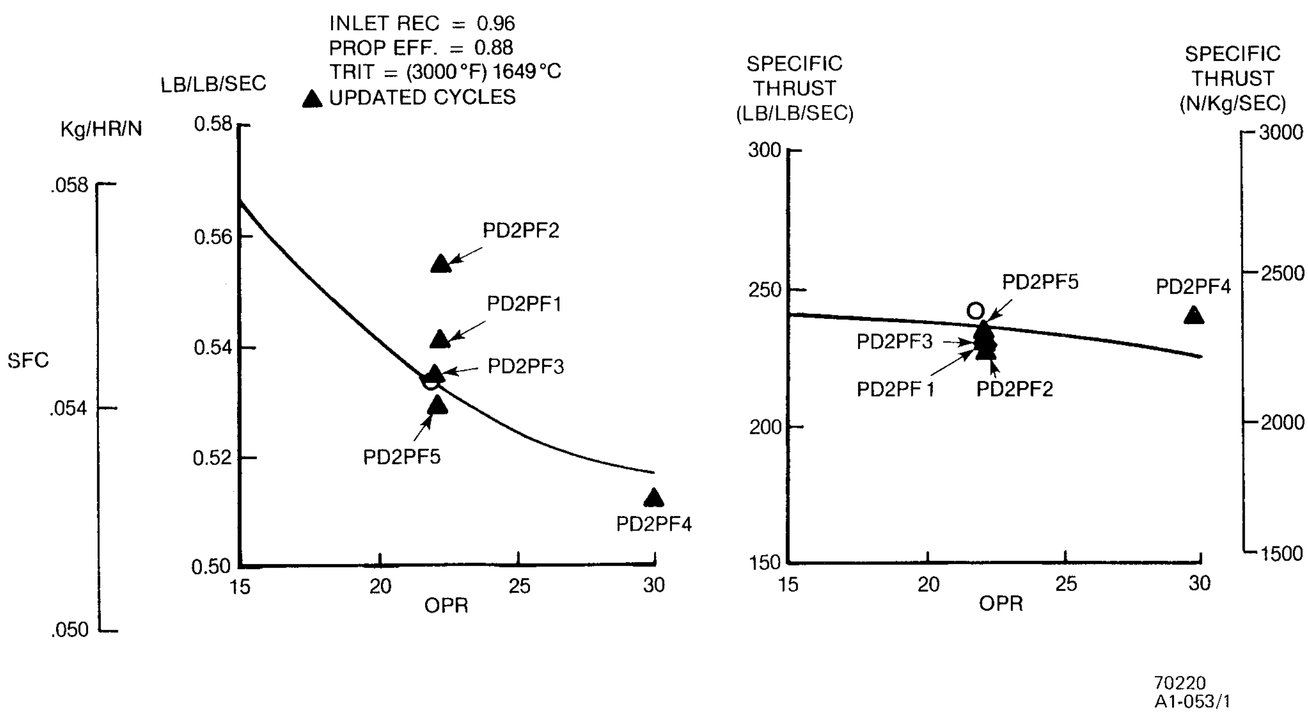

Figure 47. Two-Spool Propfan Parametric Performance With Updated Cycles: Sea Level, Mach 0.7.

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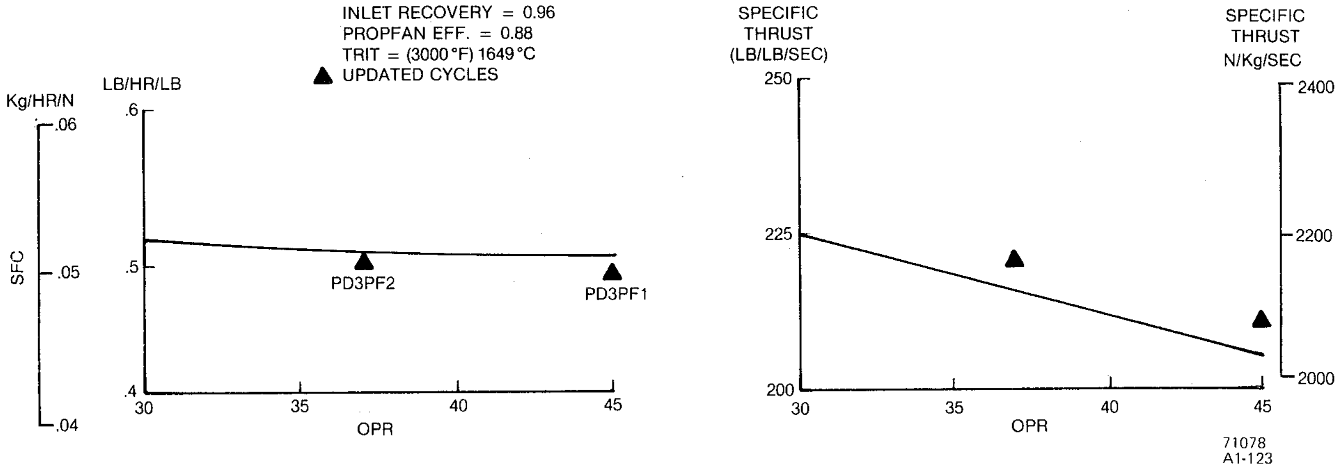
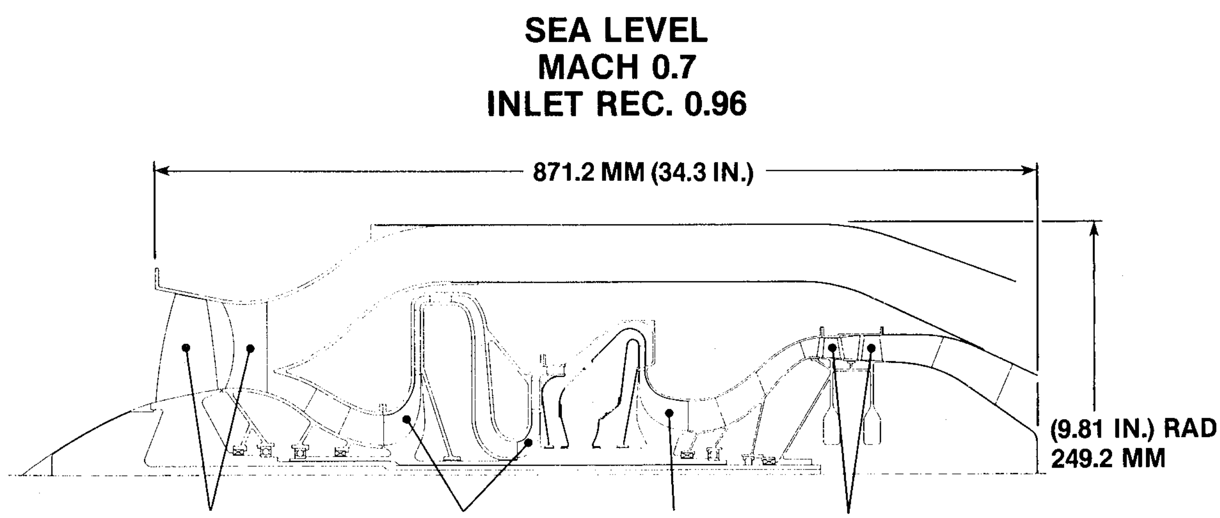


Figure 48. Three-Spool Propfan Parametric Performance With Updated Cycles: Sea Level, Mach 0.7.



$W\sqrt{\Theta}/\delta$ Kg/SEC (LB/SEC)	17.62 (38.84)	1.66 (3.65)	0.24 (0.53)	0.91 (2.00)
PR	1.65	15.78	4.4	3.95
η	0.917	0.824	0.882	0.892
$N/\sqrt{\Theta}$(RPM)	27,420	61,755	29,760	14,110
N(RPM)	28,735	69,920	69,920	28,735

THRUST, N (LB): 3772 (848) TRIT, °C (°F): 1371 (2500)
SFC, Kg/HR/N (LB/HR/LB): .073 (0.716) OPR: 26

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Figure 49. Two-Spool Turbofan Candidate Engine Cross-Section And Performance.

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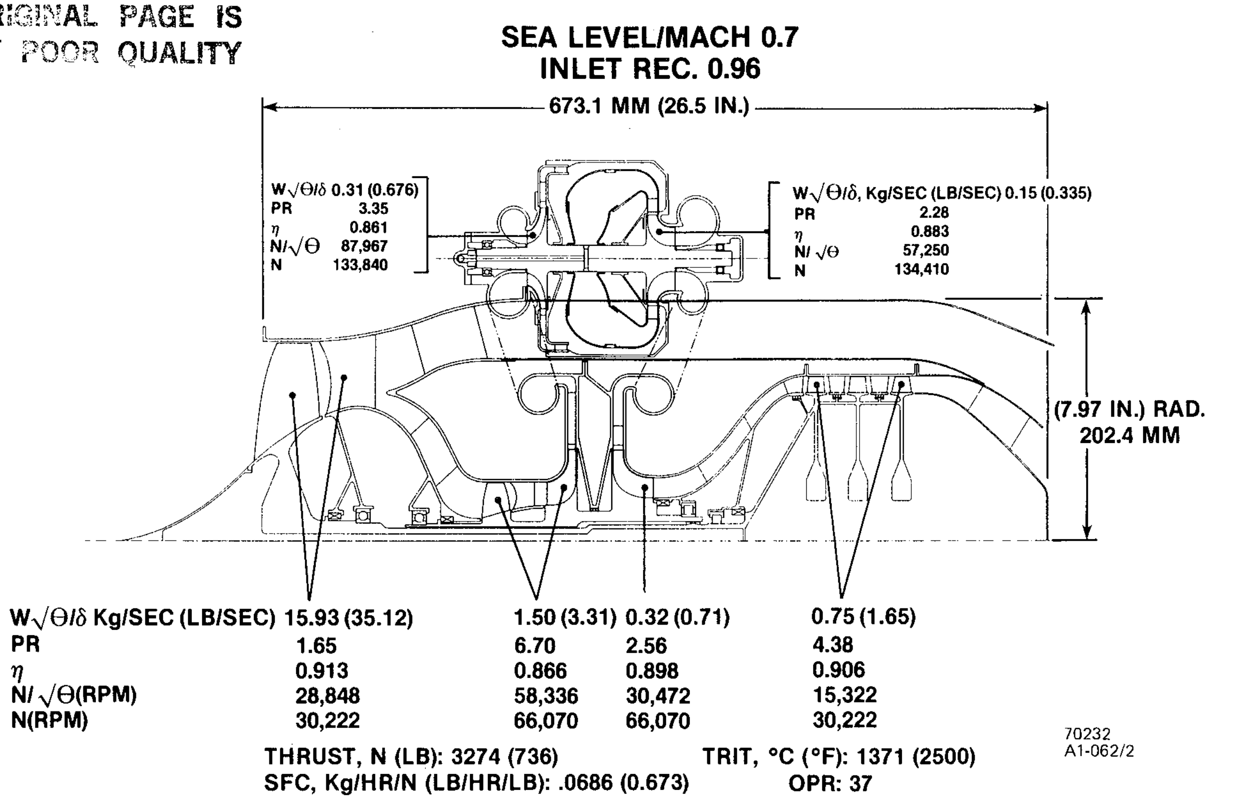

Figure 50. Three-Spool Turbofan Candidate Engine Cross-Section And Performance.

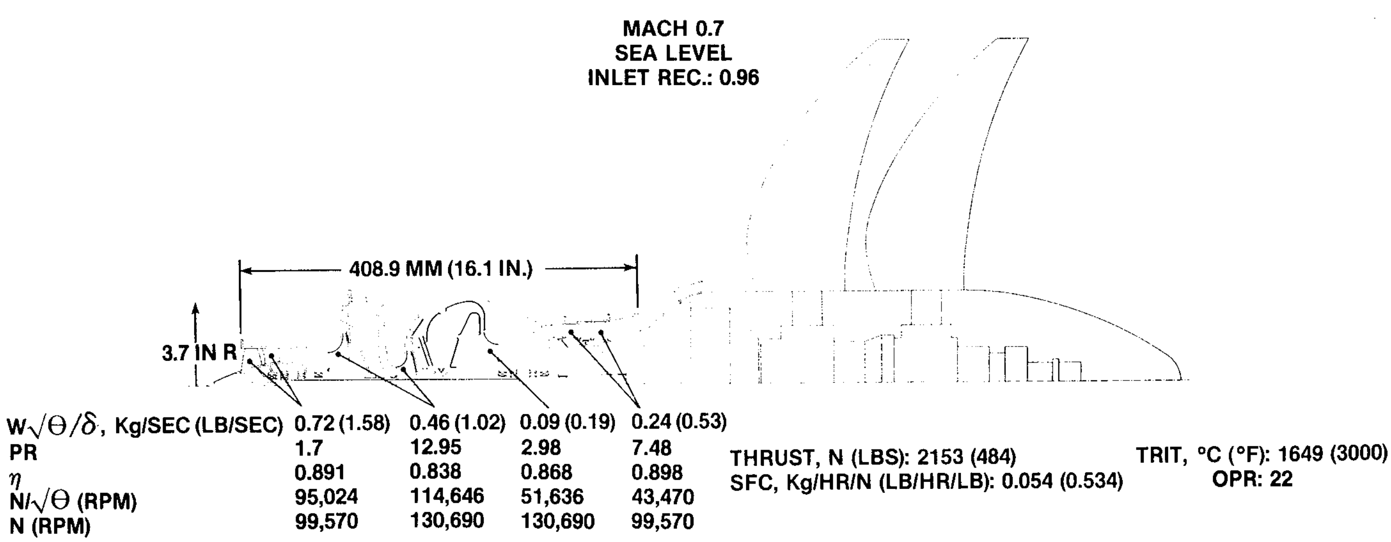

Figure 51. Two-Spool Propfan Candidate Engine Cross-Section And Performance.

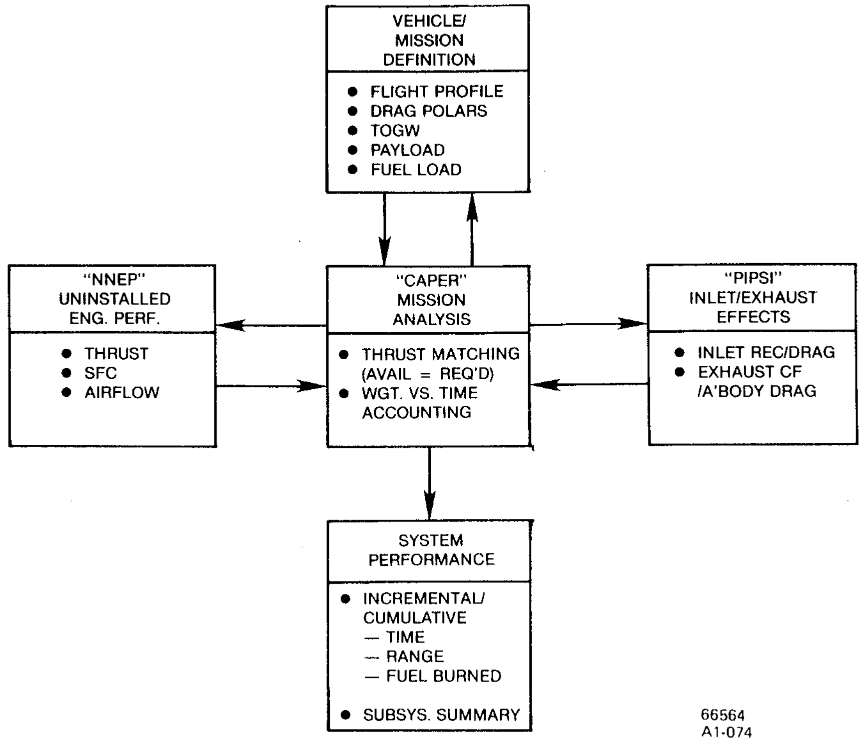

Figure 52. Mission Analysis Methodology Flow Chart.

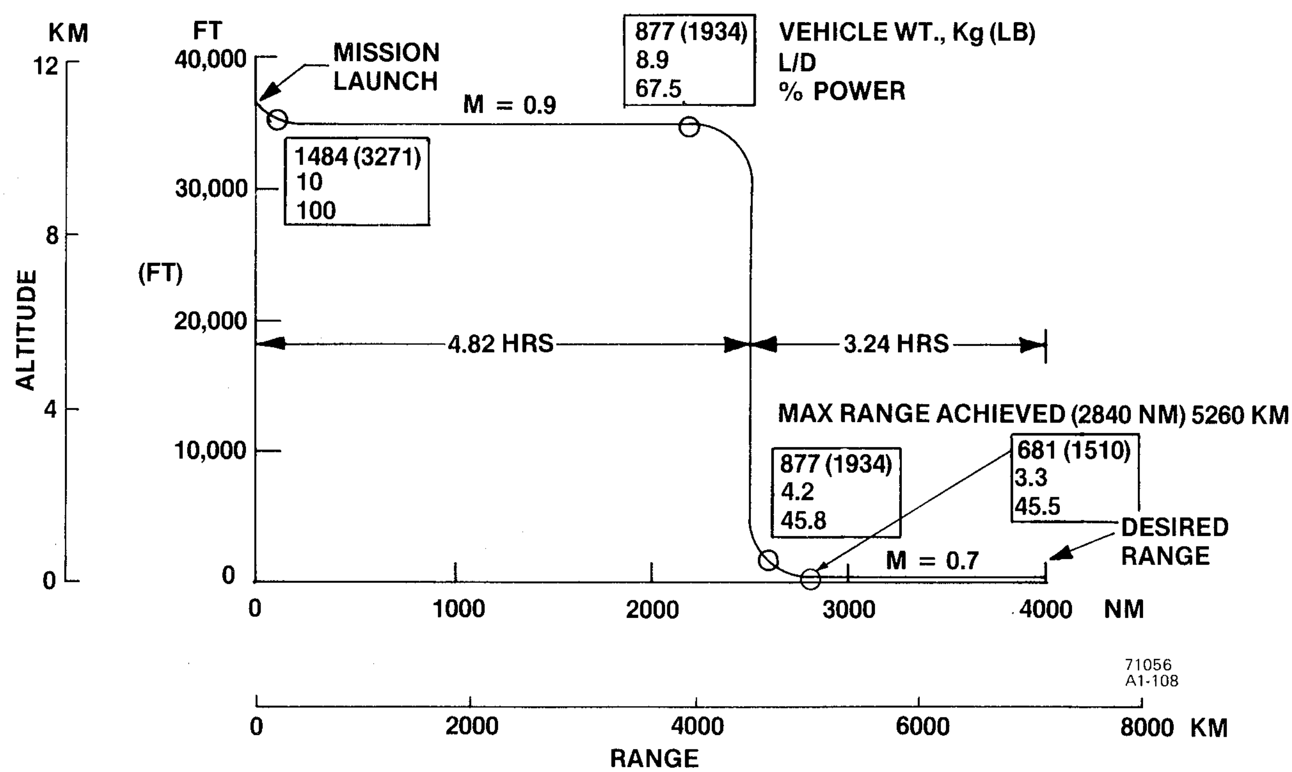

Figure 53. Cruise Missile Mission Performance: Baseline Engine.

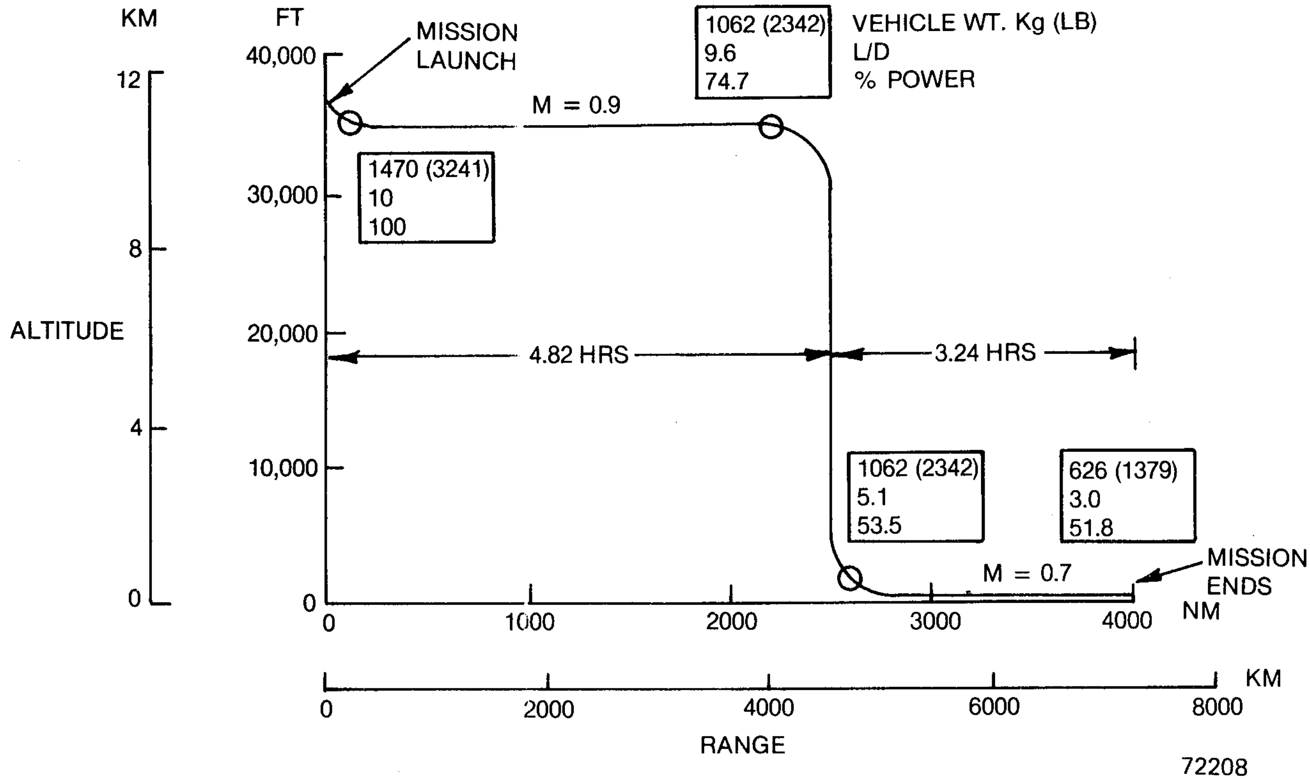

Figure 54. Cruise Missile Mission Performance: Two-Spool Advanced Technology Turbofan Engine.

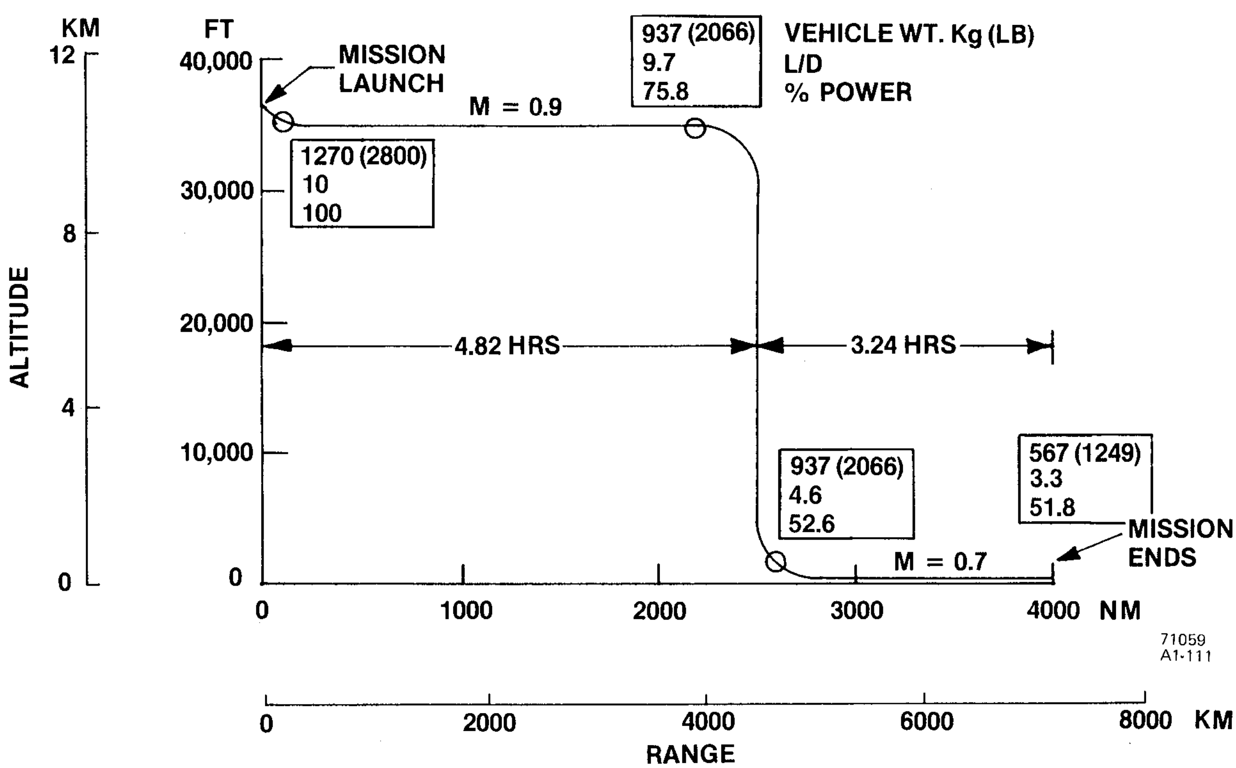

Figure 55. Cruise Missile Mission Performance: Three-Spool Advanced Technology Turbofan Engine.

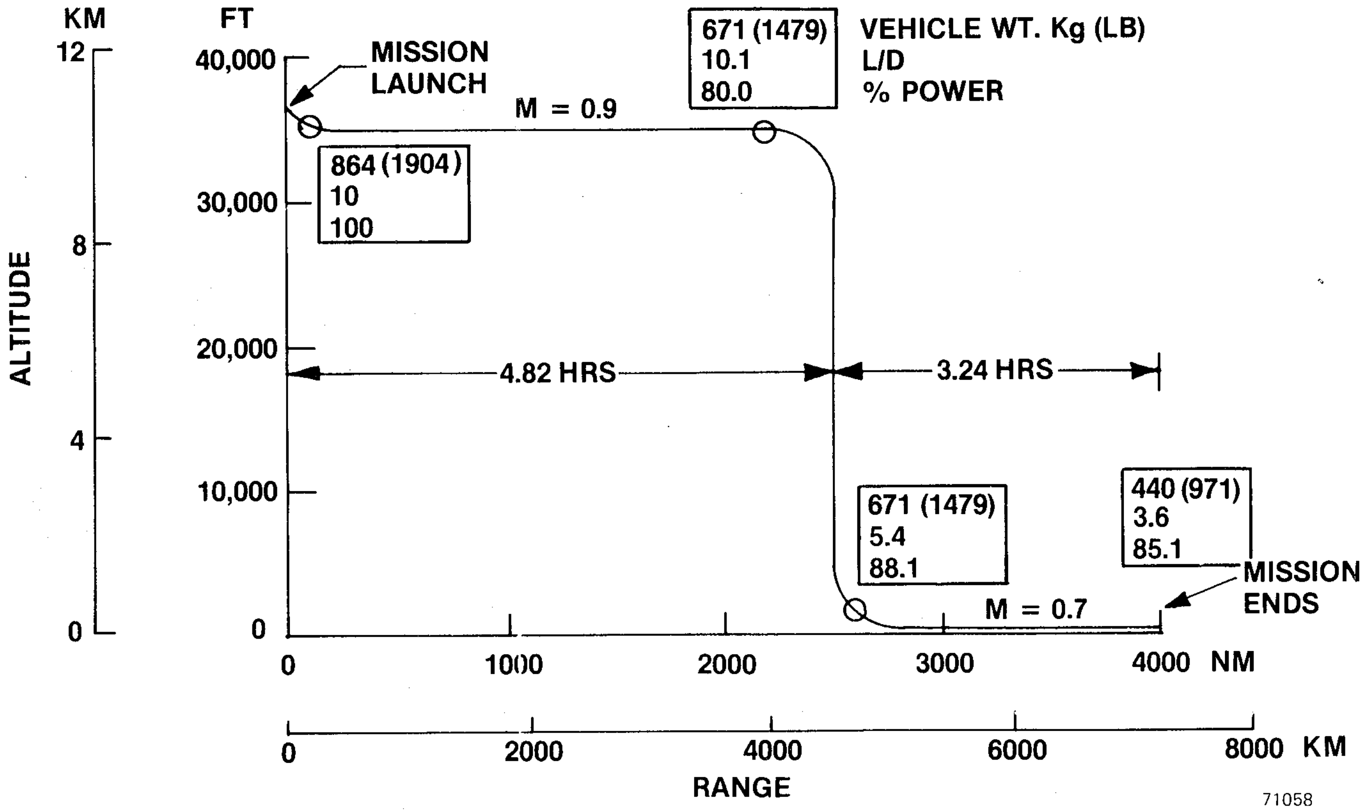

Figure 56. Cruise Missile Mission Performance: Two-Spool Advanced Technology Propfan Engine.

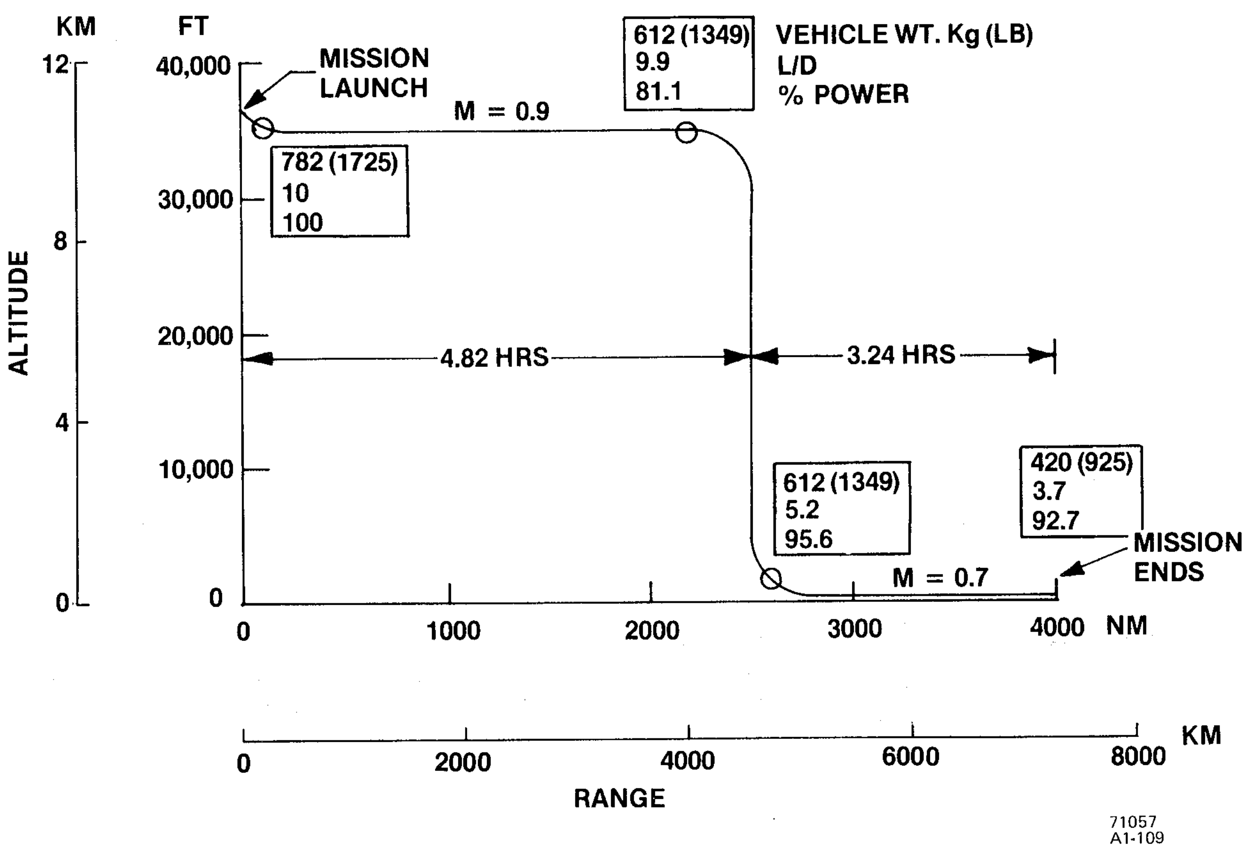

Figure 57. Cruise Missile Mission Performance: Three-Spool Advanced Technology Propfan Engine.

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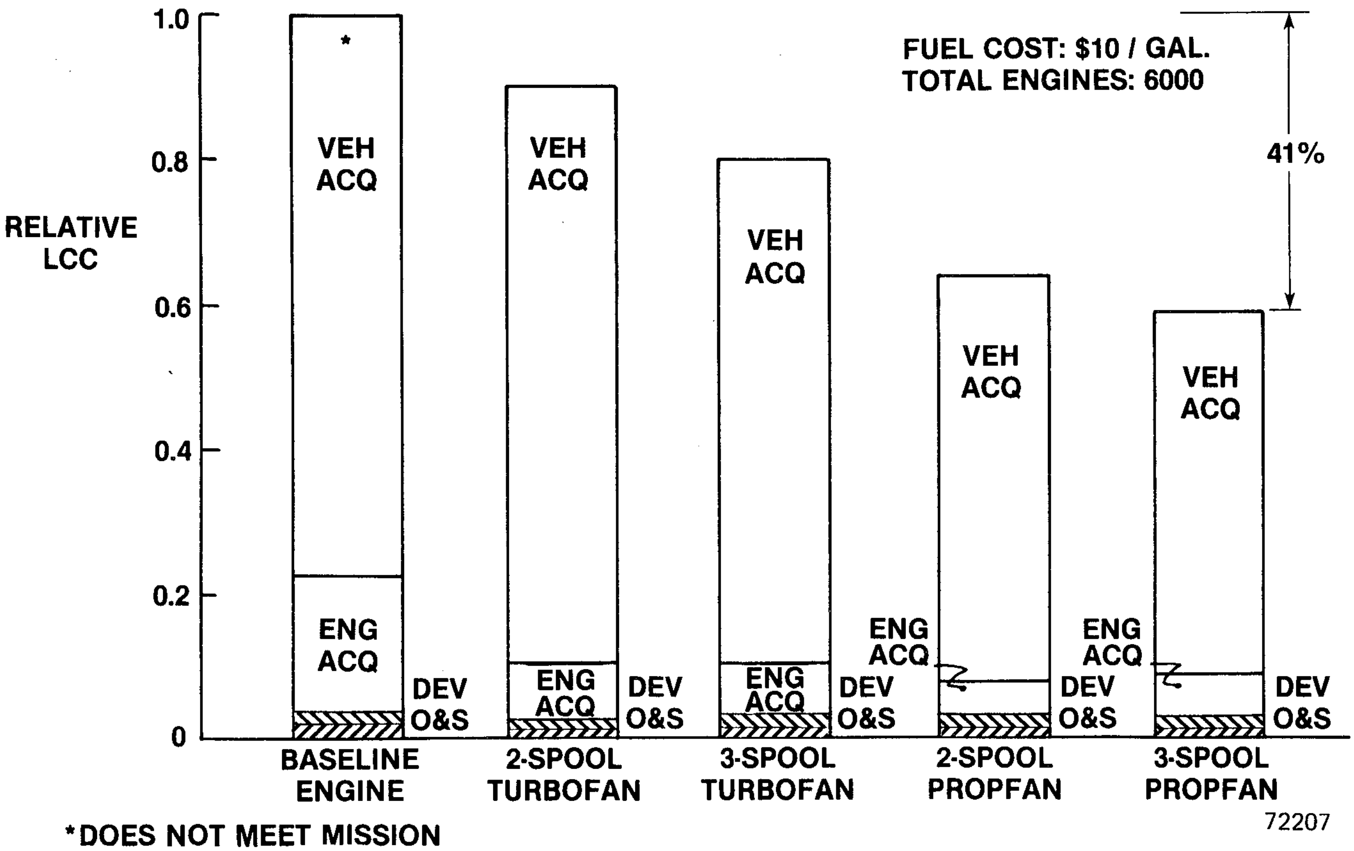


Figure 58. System Life Cycle Cost Results.

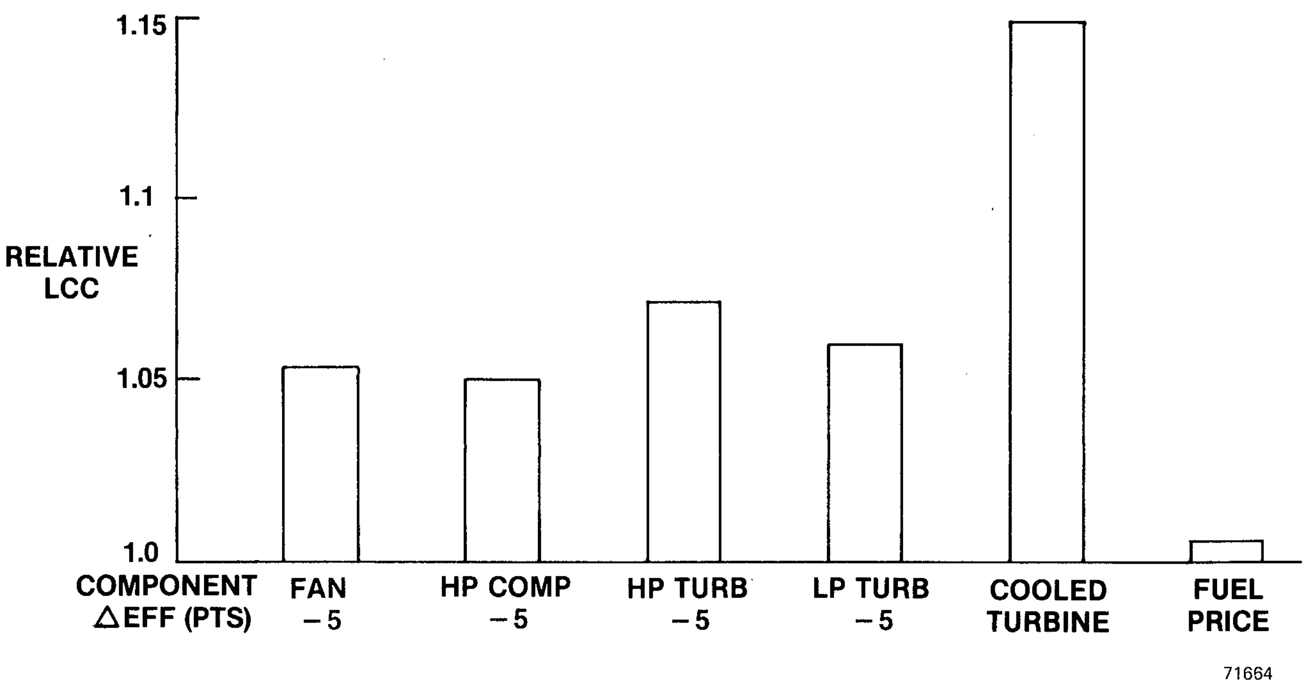
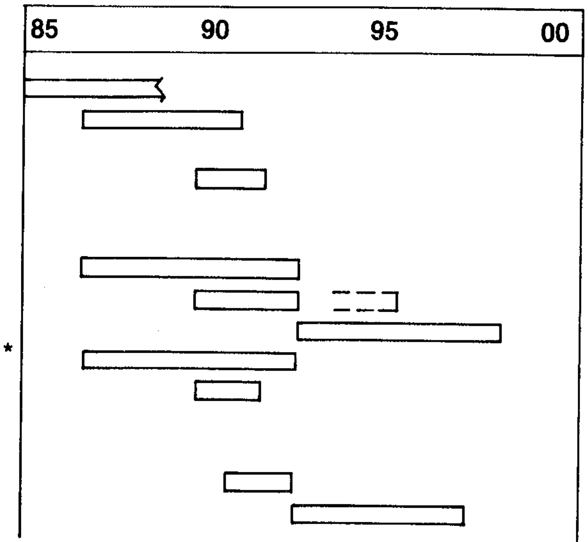


Figure 59. System Life Cycle Cost Sensitivity Results.

1. AERODYNAMIC TECHNOLOGY
   - ON-GOING R&D
   - INVERSE DESIGN METHODOLOGY*
   - PERF. VERIFICATION (COLD) TESTS
     — BENCH, SUBCOMPONENT & COMPONENT TESTS (3 SPOOL DUCT, T.I.N.)
2. COMPOSITE MAT'L/MANUF./STRUC. TECHNOLOGY
   - MAT'L DEVEL. & PROPERTY ARCHITECTURE
   - MAT'L SAMPLE/COMPONENT TESTS
   - MAT'L MANUF. DEVELOPMENT**
   - STRUCT. DESIGN METHODOLOGY/OPTIM. DES. TOOLS*
   - VERIFICATION TESTS (SAMPLE/COMPONENTS)
3. FULL SCALE SIMULATION/ENGINE TESTS
   - UPDATED/HIGH TEMP. SIMULATION TESTS
   - DEMO ENGINE TESTS



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** OUTSIDE THE SECT PROGRAM, PROVIDED FOR PLANNING ONLY

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Figure 60. Technology Program Plans: Radial Inflow Turbine.

1. HIGH TEMP. MAT'L/MANUF./STRUCT. TECHNOLOGY
   - MAT'L DEVEL. & PROPERTY ARCHITECTURE
   - MAT'L SAMPLE/COMPONENT TESTS
   - MAT'L MANUF. DEVELOPMENT*
   - BEARING COMPONENT DEVELOPMENT
   - HIGH TEMP LUBRICATION (LIQUID, DRY) DEVEL.
   - HIGH TEMP. SEAL DEVELOPMENT
   - BEARING/LUBRICATION VERIF. TESTS (BENCH, RIG)
   - SEALS STRUC. TESTS (BENCH, RIG)
2. FULL SCALE SIMULATION/ENGINE TESTS
   - DEMO ENGINE TESTS*

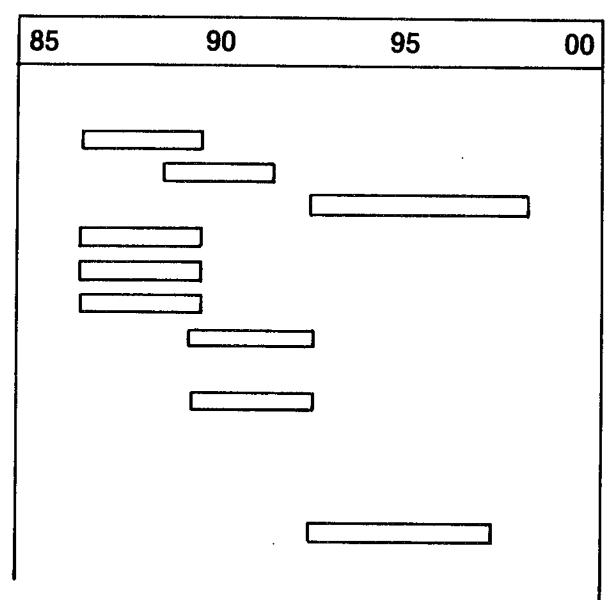

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Figure 61. Technology Program Plans: Bearings And Seals.

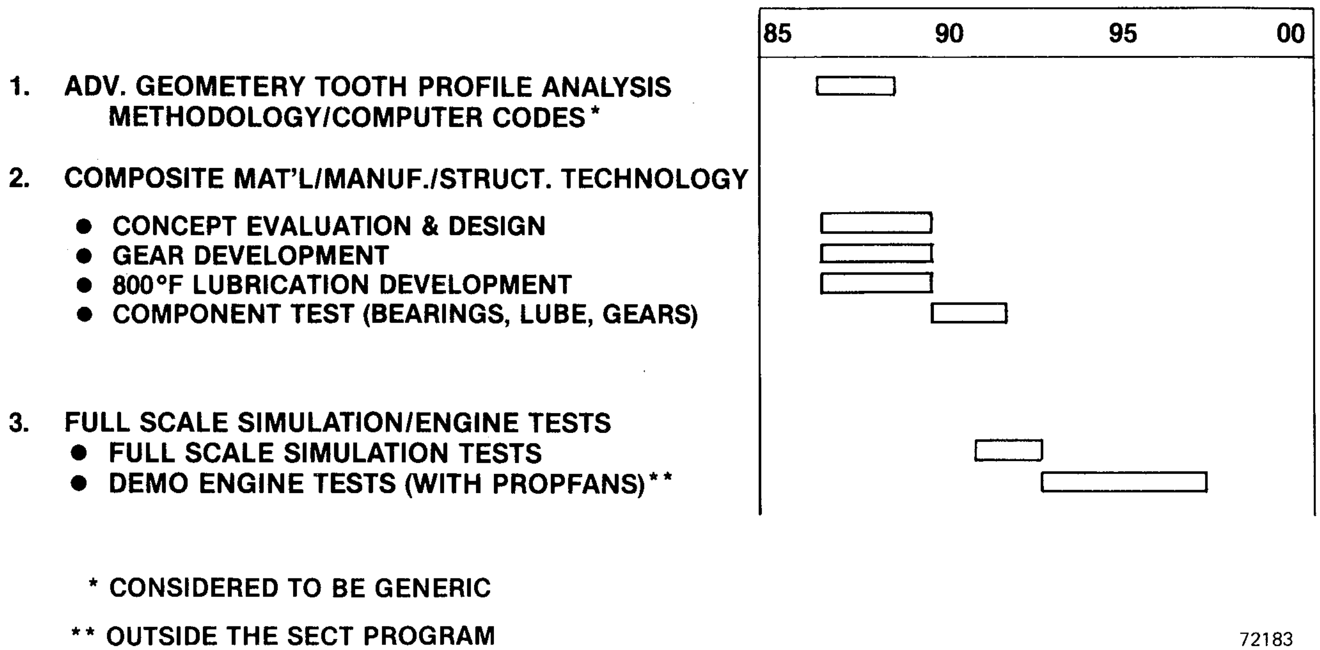

Figure 62. Technology Program Plans: Propfan Gearbox.

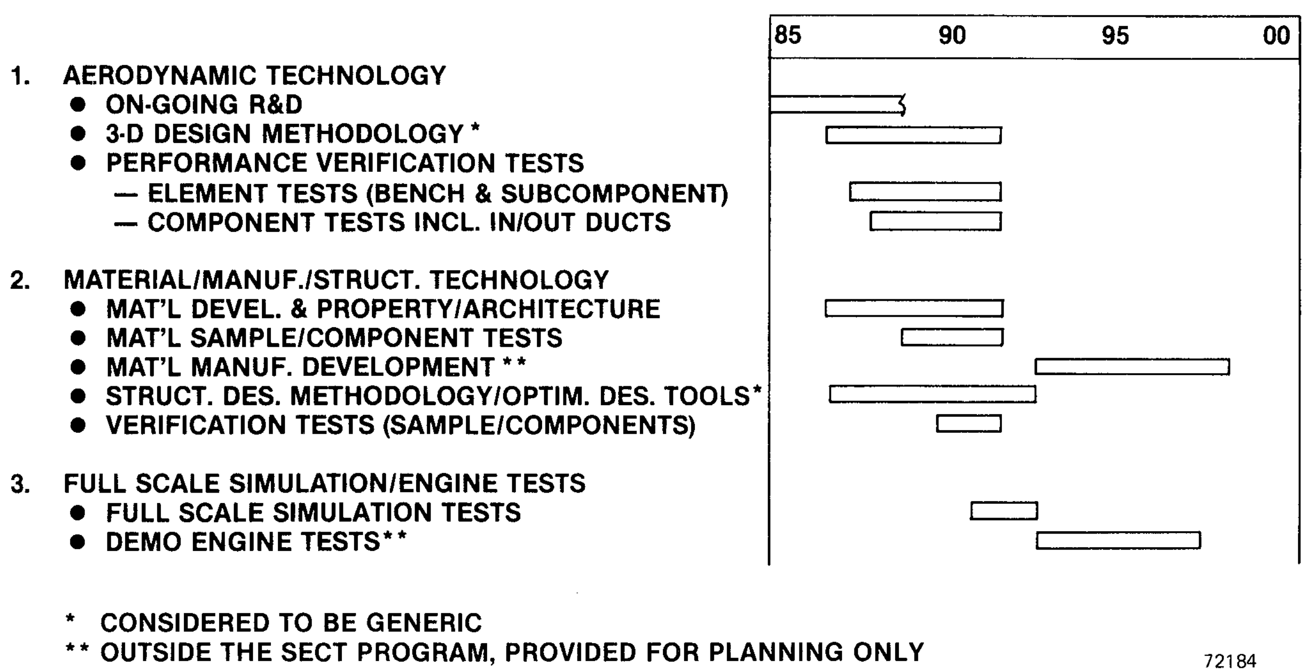

Figure 63. Technology Program Plans: Dual Centrifugal Compressor.

1. AEROTHERMODYNAMIC TECHNOLOGY
   - 3-D REACTION FLOW METHODOLOGY *
   - PERF. VERIFICATION & CHARACTERIZATION TESTS
2. COMPOSITE MAT'L/MANUF./STRUCT. TECHNOLOGY
   - MAT'L DEVEL. & PROPERTY/ARCHITECTURE
   - MAT'L SAMPLE/COMPONENT TESTS
   - MAT'L MANUF. DEVELOPMENT **
   - STRUCT. METHODOLOGY & OPTIMIZ. DES. TOOLS *
   - STRUCT. VERIFICATION TESTS (SAMPLE/COMPONENTS)
3. FULL SCALE SIMULATION/ENGINE TESTS
   - FULL SCALE SIMULATION TEST
   - DEMO ENGINE TESTS **

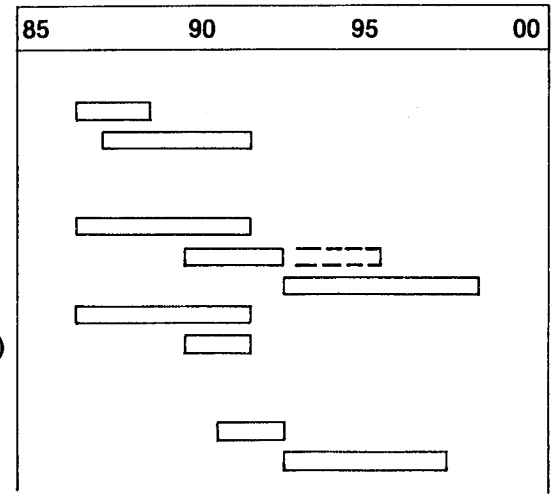

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Figure 64. Technology Program Plans: Slinger Combustor.

1. FIBER REINF. SUPERALLOYS MAT'L/MANUF./STRUT TECHNOLOGY
   - PRIOR R & D
   - MAT'L DEVEL. & PROPERTY ARCHITECTURE
   - MAT'L SAMPLE/COMPONENT TESTS
   - MAT'L MANUF. DEVELOPMENT *
   - STRUCT DESIGN/METHODOLOGY (STABILITY, HIGH TEMP. VISCOUS DAMPER)
   - COMPONENT TESTS (BENCH, RIG)
2. FULL SCALE SIMULATION/ENGINE TESTS
   - DEMO ENGINE TESTS *

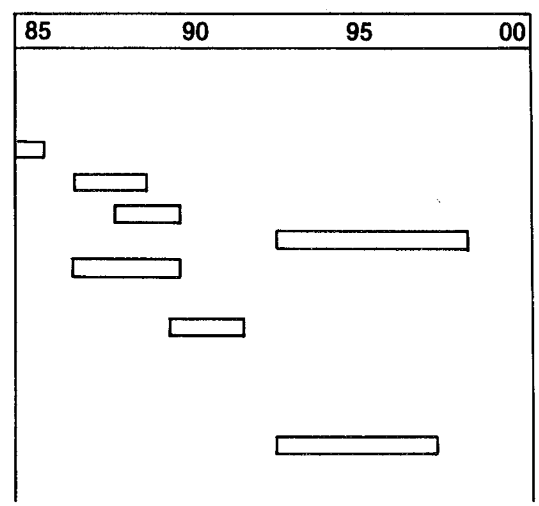

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Figure 65. Technology Program Plans: High Speed Shaft.

APPENDIX A

LIST OF SYMBOLS AND ABBREVIATIONS

SYMBOLS	DEFINITION	UNITS	
AN^2	ANNULUS AREA-SPEED STRESS PARAMETER	M^2-REV^2/Sec^2	$(IN\text{-}RPM)^2$
BPR	BYPASS RATIO	---	---
Co	ISENTROPIC VELOCITY CORRESPONDING TO TOTAL HEAD	M/SEC	(Ft/Sec.)
D	PROPELLER TIP DIAMETER	M	(FT)
D_M	SHAFT MEAN DIAMETER	MM	(IN.)
F_N	NET THRUST	Newton	(LB.)
G	ACCELERATION DUE TO GRAVITY	MM/Sec^2	(FT/Sec^2)
Had	TOTAL TO STATIC ADIABATIC HEAD	M	(FT.)
HCF	HIGH CYCLE FATIGUE	---	---
HP	HIGH PRESSURE	---	---
I	COMBUSTOR HEAT RELEASE RATE	J/SEC. M^3 Pa	$(MBTU/HR\text{-}FT^3\text{-}ATM)$
IP	INTERMEDIATE PRESSURE	---	---
J	ENERGY CONVERSION FACTOR	JOULE	(FT-LB/BTU)
KD2	INLET DISTORTION INDEX $= \dfrac{\sum_{ID}^{OD}\left[(\frac{\Delta P}{P})_{RING} \times (\Theta^{-})_{RING} \times C_{RING}\right]}{\sum_{ID}^{OD} C_{RING}}$	---	---
L	SHAFT LENGTH	M	(IN.)
LCC	LIFE CYCLE COST	---	---
LCF	LOW CYCLE FATIGUE	---	---
M	MACH NUMBER	---	---
M_2	EXHAUST RELATIVE MACH NUMBER	---	---
N	SPEED	RPM	(RPM)
n	SPEED	Rev/Sec	(Rev/Sec)
N_S	SPECIFIC SPEED $\left[\dfrac{n\sqrt{\Theta}}{[Had\ 3/4]}\right]$	---	---
OPR	OVERALL PRESSURE RATIO	---	---
P_3	COMBUSTOR INLET PRESSURE	Pa	(LB/IN^2)
PR	PRESSURE RATIO	---	---
Q	VOLUME FLOW EXIT OF TURBINE	M^3/Sec	$(FT^3/Sec.)$
R_H	HUB RADIUS	MM	(IN.)
R_T	TIP RADIUS	MM	(IN.)
SFC	SPECIFIC FUEL CONSUMPTION	Kg/HR/N	(LB/HR/LB)
SHP	SHAFT HORSEPOWER	KW	(HP)
SPECIFIC THRUST	THRUST/AIRFLOW	N/Kg/Sec	(LB/LB/Sec.)
SPECIFIC VOLUME	COMBUSTOR SPECIFIC VOLUME [Volume/ $\dfrac{W3\sqrt{T3}}{P_3}$/]	M^3 - $SEC/MM^2\sqrt{{}^oK}$	$(FT^3\text{-}Sec./In^2\text{-}\ {}^oR)$
T_3	COMBUSTOR INLET TEMPERATURE	^oK	(^oR)
T_4	COMBUSTOR EXIT TEMPERATURE	^oK	(^oR)
T_C	COOLING BLEED TEMPERATURE	^oK	(^oR)
T_g	GAS TEMPERATURE	^oK	(^oR)
T_W	METAL TEMPERATURE	^oK	(^oR)

LIST OF SYMBOLS AND ABBREVIATIONS (CON'T)

SYMBOLS	DEFINITION	UNITS	
TEMPERATURE COEFFICIENT	$GJ\Delta H/U_T^2$	---	---
TRIT	TURBINE ROTOR INLET TEMPERATURE	°C	(°F)
U	MEAN WHEEL SPEED	M/SEC	(FT/Sec.)
U_T	TIP SPEED	M/SEC	(FT/Sec.)
W	VEHICLE WEIGHT	Kg	(LBS)
W_3	COMBUSTOR INLET AIRFLOW	Kg/SEC	(LB/Sec.)
W_E	ENGINE WEIGHT	Kg	(LBS)
WA	AIRFLOW	Kg/SEC	(LB/Sec.)
γ	SPECIFIC HEAT RATIO	---	---
ΔH	ENTHALPY CHANGE	J/g	(BTU/LB)
$\Delta\beta_{HUB}$	GAS TURNING ANGLE	Degrees	(Degrees)
δ	RATIO OF PRESSURE TO SEA LEVEL STANDARD	---	---
ϵ	FUNCTION OF $\gamma = \frac{\gamma_s}{\gamma}\left[\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma-1}{\gamma}}\left(\frac{\gamma_s+1}{2}\right)^{\gamma_s/\gamma_s-1}\right]$	---	---
E_C	COOLING EFFECTIVENESS $\frac{(Tg - Tw)}{Tg - Tc}$	---	---
η	ADIABATIC EFFICIENCY		---
η_P	POLYTROPIC EFFICIENCY	---	---
Θ	RATIO OF TEMPERATURE TO SEA LEVEL STANDARD	---	---
Θ_{cr}	RATIO OF CRITICAL VELOCITY AT TURBINE INLET TEMPERATURE TO CRITICAL VELOCITY AT SEA LEVEL STANDARD		---
ρ_E	ENGINE DENSITY	Kg/M^3	(LB/FT^3)
τ	RESIDENCE TIME	Millisecond	(Millisecond)
$\frac{\partial W}{\partial (Fn/W_E)}$	RATIO OF CHANGE OF VEHICLE WEIGHT WITH RESPECT TO ENGINE THRUST TO WEIGHT RATIO	---	---
$\frac{\partial W}{\partial SFC}$	RATIO OF CHANGE OF VEHICLE WEIGHT WITH RESPECT TO SFC	---	---
$\frac{\partial W}{\partial \rho_E}$	RATIO OF CHANGE OF VEHICLE WEIGHT WITH RESPECT TO ENGINE DENSITY	---	---

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