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SMALL ENGINE COMPONENT TECHNOLOGY (SECT) STUDY

FINAL REPORT

BY

P.K. MEYER AND L. HARBOUR

WILLIAMS INTERNATIONAL

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Project Manager, Michael R. Vanco
NASA Lewis Research Center
Cleveland, OH 44135

16. Abstract

A study was conducted to identify component technology requirements for small, expendable gas turbine engines that would result in substantial improvements in performance and cost by the year 2000. A subsonic, 2600 nautical mile (4815 kilometers) strategic cruise missile mission was selected for study. A baseline (state-of-the-art) engine and missile configuration were defined to evaluate the advanced technology engines. Two advanced technology engines were configured and evaluated using advanced component efficiencies and ceramic composite materials: a 22:1 overall pressure ratio, 3.85 bypass ratio twin-spool turbofan; and an 8:1 overall pressure, 3.66 bypass ratio, single-spool recuperated turbofan with 0.85 recuperator effectiveness. Results of mission analysis indicated a reduction in fuel burn of 38% and 47% compared to the baseline engine when using the advanced turbofan and recuperated turbofan, respectively. While use of either advanced engine resulted in approximately a 25% reduction in missile size, the unit life cycle cost (LCC) reduction of 56% for the advanced turbofan relative to the baseline engine gave it a decisive advantage over the recuperated turbofan with 47% LCC reduction. An additional range improvement of 10% results when using a 56% loaded carbon slurry fuel with either engine. These results can be realized only if significant progress is attained in the fields of solid lubricated bearings, small aerodynamic component performance, composite ceramic materials and integration of slurry fuels. A technology plan outlining prospective programs in these fields is presented.

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PREFACE

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M. J. Bak (Slurry Fuels, Combustors)

G. C. Clark (Project Manager)

G. S. Cruzen (Advanced Materials)

T. Galvin (Life Cycle Cost)

R. A. Horn, Jr. (Manager - Aerothermodynamics)

D. S. Musgrave (Small Component Aerodynamics)

J. D. Paletta (Bearings)

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SUMMARY

Williams International has conducted studies to identify component technology requirements for substantial performance and cost improvements applicable to subsonic, strategic, cruise missile engines for the year 2000.

In the first phase a 2600 nautical-mile (4815 km) reference mission for the year 2000 was formulated. A current state-of-the-art engine was selected as a reference engine to which advanced engines were compared. A reference airframe was defined which was used in conjunction with the reference engine and also the advanced engines. Definition of the evaluation procedure and groundrules for the remaining phases of the study completed the first phase.

Advanced engine thermodynamic cycle configuration evaluation was carried out in the second phase of the study. It resulted in the formulation of two advanced technology engines, each capable of performing the reference mission with the reference airframe. They feature high efficiency aerodynamic components, solid lubricated bearings, and ceramic composite high pressure turbine. The design point characteristics of these engines are compared to the reference engine as follows:

	Reference Engine	Advanced Turbofan	Recuperated Turbofan
Number of Spools	2	2	1
Bypass Ratio	1.02	3.85	3.66
Overall pressure ratio	13.6:1	22:1	8:1
TIT: ^{o}F (^{o}C)	1970(1077)	2200(1204)	2650(1454)
Recuperator effectiveness	N/A	N/A	0.85
SFC: lbm/lb-hr (kg/N-hr)	0.987(0.101)	0.765(0.078)	0.6663(0.0680)

System performance evaluation using these three engines was carried out in the third phase. The following results are relative to the reference engine:

	Advanced Turbofan	Recuperated Turbofan
Reduction in Fuel Burn	38%	47%
Reduction in Missile Size	22%	25%
Reduction in Life Cycle Cost (LCC)	56%	47%

In addition, it was determined that range can be extended for either advanced engine by 10 percent by using a 56 percent loaded carbon slurry fuel.

Since system performance was equivalent for the two advanced technology engines and there was only a minor difference in missile size, the advanced turbofan engine was selected on the basis of significantly lower LCC.

In the final phase of the study, technology plans were prepared to outline programs required to provide technology advances needed to realize these performance and LCC gains. Ranked in order of their contribution to LCC reduction, they are:

Solid-Lubricated Bearing Program
Advanced Small Component Aerodynamics Program
Ceramic Composite Materials Program
Slurry Fuel Technology Program

1.0 INTRODUCTION

Small gas turbine engines have played an important role, over the last 30 years, in providing propulsion and power systems for a wide range of commercial and military aircraft, missiles, and remotely piloted vehicles.

For a number of these applications they have provided an efficient, minimum weight, low cost, and highly reliable solution to system requirements. Small gas turbines have not, however, received continuous and consistent application of resources to achieve maximum component and overall system performance.

Small gas turbine engines are not simply reduced size versions of larger gas turbine engines. Design requirements of their unique applications and their manufacturer's engine philosophy produce component and configuration characteristics distinctly different from those of large engines. Practical considerations, along with size effects, cause the efficiencies of small components to be several points less than larger component efficiencies. These differences are the primary reasons that large gas turbine engines exhibit better performance than the smaller versions.

Large engine design and manufacturing techniques, while enlarging the fundamental understanding of gas turbine design, do not address the gamut of particulars for the small gas turbine. Therefore, this technology is not entirely transferable to small engine design. Despite limited resources, the small engine manufacturers have made notable advances in small gas turbine performance, but a concerted and coordinated research effort to enhance component technology offers significant benefit for improving small turbine engine performance characteristics.

Technological progress is a direct result of the potential payoffs, cost of research, and the inherent risks. In the field of small gas turbines these payoffs usually include reduced cost, improved reliability, and increased performance. The relative importance of these payoffs depend, of course, on application. Additionally, missile applications are often concerned with improved survivability and storage life. Criteria that dictate successful engine design for a subsonic strategic cruise missile, for example, depend on the success criteria that drive the missile system design. These, in turn, depend on the mission success criteria that include:

- Avoid detection,
- Avoid defenses,
- Confuse defenses,

- Expand number of targets, and

- Protect the launch platform.

These payoff goals have dictated continuing efforts to achieve design simplicity, improved energy utilization, improved fabrication techniques and additionally, for missile engines, environmentally stable fuels, lubricants, and sealants.

There are many areas where advancements in aerodynamics, materials, and component technology would benefit future engines. This study, sponsored by NASA Lewis Research Center and U.S. Army Aviation Research and Technology Activity - Propulsion Directorate, documents those areas of highest potential payoff by the year 2000 for subsonic strategic cruise missile engines. The study was undertaken in four distinct phases.

- Task I — Select evaluation procedures and assumptions that will govern the study. Select a reference mission, a reference airframe, and a reference engine to be used in the evaluation.

- Task II — Evaluate applicable gas turbine engine cycles and configurations and define two of the most promising advanced technology engines.

- Task III — Conduct mission and cost analyses to evaluate the payoff realized from the advanced technology.

- Task IV — Provide a technology plan that will result in achieving the required advancement in technology.

Symbols and abbreviations used in this report are listed in Appendix A.

2.0 TASK I - SELECTION OF EVALUATION PROCEDURES AND ASSUMPTIONS

The importance of achieving a high degree of system survivability for cruise missiles has placed significant emphasis on the design of engines with a balance between high specific thrust and low specific fuel consumption. The Small Engine Component Technology (SECT) Program is concerned with identification of system payoffs capable of being achieved if vigorous research and development programs are pursued. The first task in the identification of system payoffs is to select the assumptions and the procedure by which engine advances will be evaluated in terms of system payoffs. This task was divided into six major elements:

- Definition of current cruise missile launch modes,
- Identification of reference year 2000 missions,
- Definition of reference year 2000 airframe,
- Definition of state-of-the-art reference engine, and
- Definition of evaluation procedures and assumptions.

The approach taken was to first define the current cruise missile missions. Factors that are expected to redefine those missions were then examined as well as the factors that will contribute to mission success. These were combined to formulate mission characteristics representative of the year 2000 subsonic strategic cruise missile. A list of reference material is included at the end of this report.

2.1 Current Cruise Missile Missions

The launch mode for cruise missiles places them in three categories:

- Air-launched cruise missiles (ALCM),
- Ground-launched cruise missiles (GLCM), and
- Sea-launched cruise missiles (SLCM).

The SLCM may be launched from either a submerged submarine or from a surface ship. Once launched, the basic operational mode for each one is essentially the same.

An optimum altitude cruise may be used in order to provide longer range capability. Cruise Mach number is subsonic. Mission length utilizing a low level cruise is in the range of 1350 to 1700 nautical miles (2500 to 3148 km). Minimum stand-off is indicated to be 300 nautical miles (556 km) and maximum stand-off is 800 nautical miles (1482 km).

2.2 Factors That Will Contribute To Mission Success

The following factors must be optimized:

- Avoid detection,
- Avoid defenses,
- Confuse defenses,

- Expand the number of targets vulnerable to attack, and
- Protect the launch platform.

2.3 Reference Year 2000 Mission

Reference year 2000 missions were formulated to provide a means of evaluating advanced technology engines in comparison to a reference state-of-the-art engine in a reference year 2000 airframe. The launch mode is basically either air-launch (ALCM) or surface-launch (GLCM or SLCM). A range of 2600 nautical miles (4815 km) for the air-launched strategic cruise missile was selected. The first 500 to 1000 nautical miles (926 to 1852 km) may be flown at an altitude for optimum cruise. The greater stand-off range is required to protect the carrier. The four reference missions selected are shown in Figure 1. The stand-off range of 600 nautical miles (1111 km) was chosen because it represents an average stand-off range of the four launch modes. Upon completion of the stand-off phase, the missile flies a subsonic terrain following mode which is the same for all four reference missions. Included in this mode are two subsonic dash segments. A final dash segment to the target completes the mission.

2.4 Reference Year 2000 Airframe

In the year 2000, the cruise missile airframe will be configured to extend the range of the cruise missile. Some compromise will most certainly be required since it is not likely that improvements in engine efficiency and high energy/high density fuels alone can provide the desired increase in range.

For the most efficient cruise, the cruise wing loading should produce near-maximum wing lift-drag ratio and this occurs near maximum lift coefficient. Thus, another conflict occurs, where aerodynamic efficiency will limit the maneuverability of the cruise missile.

Several objectives and ground rules were qualitatively formulated to guide the selection.

A. Missile diameter of 20.38 inches (51.8 cm) was selected to be compatible with current configurations.

B. The missile has a flush inlet. This posed two challenges to the engine.

   1. The SFC penalty due to the increased inlet loss had to be made up by the engine before any net SFC improvement could be realized.

2. The compressor components had to be able to tolerate the increased flow distortion characteristic of a flush inlet, especially at negative angle of attack.

C. Aerodynamic characteristics were those of existing missiles with the following two exceptions:

1. It was postulated that drag coefficient at high subsonic Mach numbers will increase less rapidly due to a finer nose shape.

2. Variable camber airfoil (leading edge slats, flaps) will provide temporary high lift capability for extreme maneuver capabilities.

D. By the year 2000, weight of the missile structure and components (less engine) will be reduced through the use of composites and further component miniaturization.

E. Airframe Shape

The year 2000 cruise missile airframe shape is shown in Figure 2. Blending of the wing and fins with the body will result in a "cusped" body cross-sectional shape shown in Figure 3 to improve aerodynamic efficiency.

F. Missile Size

Missile size is ultimately driven by the volume of fuel required to complete the 2600-nautical mile (4815 km) reference mission using the air-launched/optimum cruise altitude mode. Engine length will also be a factor. The length of the missile occupied by other components will remain fixed regardless of the engine used.

2.4.1 Reference Missile Aerodynamics

The reference missile aerodynamics are based on the current drag polars but assume no change in C_{DO} with Mach number within the missile operating range. This is justified because the transonic drag rise evident for the relatively blunt nose shape of the current shape is delayed by the sharper nose configuration of the SECT reference missile. The reference missile wing will be sized for wing loading of 214 lb/ft^2 (10246 N/m^2).

An AIAA paper (Reference 1) provides information on the effect of changing wing size on the current configurations. This information was used to determine the change in C_{DO}.

The zero lift drag coefficient for the SECT Reference Missile is a constant 0.033. The change in C_{DO} when changing wing size is assumed to be:

$$C_{DO} = (0.033 + 0.0006 \text{ (wing area-12)}) \text{ (12/wing area)}$$

based on the "wing area" used in this equation. The change in drag with changes in missile length is neglected in this study.

2.4.2 Reference Missile Physical Characteristics

The SECT reference missile physical characteristics were estimated as follows.

Engine Weight

The installed weight of the reference propulsion system was assumed to be 214 lb (97.1 kg). This includes the weight of fluids (oil and fuel), the fuel management system, and the inlet weight. The reference engine weight is 165 lb (74.8 kg).

Fixed Weight

Among the items that would not be expected to be affected by changes in missile fuel load, wing size, or body length are: residual oil and fuel, payload, wing support structure, fin panels, fin support structure, guidance section, control system, electrical system, and pneumatic system. The fixed weight is assumed to be 580 lb (263.1 kg).

Airframe Weight (Including Wings)

The airframe (body) structural weight is expected to be reduced through the use of composites. The wing weight assumes the same magnitude of weight reduction through the use of composites. The airframe weight (body plus wings) of arbitrary body length (no change in diameter) and wing area can be then expressed as:

$$\text{Airframe Weight} = 24.8 \text{ (Missile Length)} + 1.1547\text{(Wing Area)}^{1.5}$$

This relationship is used in this study to account for the change in airframe weight due to change in fuel weight (as it affects missile length) and change in wing size required to carry the resulting gross weight.

Missile Length

The missile length used in the airframe weight equation above for the year 2000 missile is estimated as follows. The length of the missile used for storing fuel is shown in Figure 4. In this length, fuel shares the space with structure and components. Assuming the missile inside diameter is 20 inches (51 cm), the minimum required length to hold fuel ($(L_{Fuel})_{Min.}$) is:

$$(L_{Fuel})_{Min} = \frac{\text{Fuel Weight}}{126.54}$$

Therefore, the fixed length is equal to the sum of hardware length, engine length, and shared length. For the SECT study, the sum of the hardware length and shared length was assumed constant. The reference missile length as a function of fuel weight is given as:

Missile Length = fixed length + fuel weight/126.54

Gross Weight

The gross weight of the reference missile consists of:

- Airframe weight as given above (W_{AF}),
- Propulsion system weight (W_{PROP}),
- Fixed weight as given above (W_{FIX}), and
- Fuel Weight (W_{FUEL})

Thus,

$$W_{GROSS} = W_{AF} + W_{PROP} + W_{FIX} + W_{FUEL}$$

This relationship is a function of engine weight and length, fuel weight, and wing area.

Note that for a given maximum wing loading (as is the case in this study), the gross weight must be arrived at iteratively since the wing size is then a function of gross weight.

2.5 Reference Engine Characteristics

A reference engine is based on a current state-of-the-art engine. The reference engine weighs 165 pounds (74.8 kg), and its characteristics at sea level, standard day conditions are given in Table I.

The reference engine is a twin spool, axial flow, turbofan engine with a single fixed geometry convergent jet nozzle. The low pressure compressor consists of a two-stage axial fan, followed by two axial compressor stages. The high pressure compressor is a single-stage centrifugal rotor. The engine utilizes an annular burner with rotary fuel injection. The high pressure turbine is a single-stage axial turbine, and is followed by a two-stage low pressure turbine. The two engine shafts are counter-rotating to minimize gyroscopic and shaft vibration effects. Engine materials are listed in Table II, and the component pressure ratios and efficiencies at the sea level operating point are listed in Table III.

The flush inlet pressure recovery characteristics shown in Figure 5 are used in all installed part load engine simulations for this study. A fuel heating value of 18,100 Btu/lb (42,100 kJ/kg) is used which reflects the SECT reference fuel (JP-10).

2.6 Evaluation Procedure And Ground Rules

The evaluation procedure recommended by Williams International to be used in the identification of component technology requirements for year 2000 cruise missile engines is presented diagrammatically in Figure 6. Successful resolution of the year 2000 mission requirements depends on four technology drivers: performance, size, cost, and reliability. Gas turbine engines must be designed with the ability to satisfy these technology drivers. The inherent conflicts between these goals mean that tradeoffs which could diminish the desired benefits of a particular goal to produce the best overall concept must be resolved during the design process. While these technology drivers are not always mutually exclusive, they are interrelated and this interplay must be balanced successfully. Advanced technology gas turbine engine concepts formulated to meet the year 2000 mission requirements must be evaluated on an overall basis considering the relative importance of each contributor. Success is measured in this study by how well the reference mission requirements are met when using the advanced technology engine in the reference airframe. The benchmark for comparison is provided by the reference engine.

2.6.1 Performance

The small gas turbine engine evaluation process begins by resolving a priority among the four success contributors. For virtually all gas turbine applications--and in particular strategic military applications--performance is the single most important contributor to success. An engine that cannot perform its assigned task as required, regardless of its cost, size or reliability, will be deemed unsatisfactory. Engine performance characteristics establish the missile system's range and maneuver capability. The performance of gas turbine engines used in cruise missiles is quantified through its specific fuel consumption, specific thrust, and its transient response characteristics. Turbine temperature, pressure ratio, component efficiencies, and heat recovery considerations are the major parameters or trade factors employed in the cycle performance analysis.

How well an engine will perform the reference missions in the reference airframe is measured by mission analysis. For this study, this is accomplished using the Williams International Integrated Systems Mission Analysis Program (ISMAP). Due to the importance of mission performance, accomplishing the 2600-nautical mile (4815 km) baseline reference mission (Figure 1) in the reference airframe is a ground rule rather than an evaluation; i.e., each engine must be able to perform the baseline reference mission. In addition, each engine must provide sufficient thrust to meet the reference maneuver criterion.

2.6.2 Size

The missile size and weight will depend primarily on the fuel required to fly the required reference baseline mission with a given engine in the reference airframe. The reference airframe internal diameter available for fuel is fixed at 20 inches (51 cm) so that reducing fuel required for the mission reduces missile length.

From a thermodynamic view point, the way to reduce fuel consumption is to employ the traditional approaches of increased cycle pressure ratio, increased turbine rotor inlet temperature (TRIT), and increased component efficiencies. Unfortunately, these approaches present many practical obstacles to the small gas turbine engine designer, and result in significant aerodynamic and cost penalties when very small blade forms are required. Each of these areas has been and will continue to be the source of challenge for the future.

Increased cycle pressure ratio in small machines has once again focused attention on centrifugal compressors and radial in-flow and mixed-flow turbines. The highly complex three-dimensional flow fields that dominate the performance of these aerodynamic components must be the focus of considerable future analytical and experimental research.

The benefit of operating at higher TRIT cannot be dismissed, but neither can the problems associated with developing reliable, cost-effective internal cooling schemes for miniature turbine blades and vanes. Development of nonmetallic substitutes promises a major improvement in small engine fuel economy by both eliminating the cooling flow penalty and permitting operation at thermodynamically optimum temperatures.

Improved aerodynamic component efficiency has been a traditional objective of engine designers. Progress is usually made in small increments and has often resulted in considerable increases in fabrication cost. Steps in component performance progress are often small, but the thermodynamic payoff can be large. Programs to reduce operating clearances and leakages and develop more sophisticated airfoil sections will continue to make progress as improvements in computational fluid mechanics and experimental assessment point to the weaknesses in current design practices.

A basic approach toward improving fuel economy that is not new but has never been widely employed is regeneration or recuperation, which has had a difficult time demonstrating that it can be applied successfully. Where weight and volume are important design considerations, neither large nor small engines have been produced using regenerative cycles. The principal problems in the past have been unacceptable volume, cost, weight, leakage, and reliability. Recent advances in ceramic materials technology and innovative new geometric configurations offer promise of eliminating many of these obstacles.

2.6.3 Cost

For expendable cruise missile engines, low cost is a key consideration in the determination of the engine design. Engine cost is influenced by the level of technical risk, materials choices, and the manufacturing procedures.

The cost of raw materials and the associated fabrication techniques required for high temperature alloys has traditionally limited the small gas turbine engine's performance potential and restricted it from entering several commercial markets, notably general aviation and automotive markets. Considerable effort will continue to be expended to replace these costly and often strategic alloys with nonmetallic substitutes. In the cold section of the engine, composites, plastics, and advanced titanium alloys are candidate material substitutes. In the hot section, ceramic, coated carbon, and composite ceramic materials hold promise. The payoff in the hot section is doubly attractive, since not only would raw material cost be reduced but fabrication costs and thermodynamic performance penalties associated with intricate cooling schemes could also be eliminated.

Engine accessories account for a disproportionately large part of the initial cost of small gas turbine engines. Much of this cost is associated with reducing the inherently high speeds of these small machines so that traditionally designed pumps and alternators can be used. Significant progress has been made in the use of high-speed accessories that eliminate the need for special gearboxes. Additional efforts at reducing costs and performance penalties associated with these high-speed accessories are also anticipated.

Another area that offers considerable promise is the growing utilization of all-electronic fuel control units. By replacing hydromechanical units, electronic fuel controls have reduced costs, improved reliability, added versatility, and eliminated many packaging problems typically found on small gas turbine engines. Future efforts to combine on-board aircraft flight control computers with the engine control into one integrated unit offer additional promise.

Review of small engine requirements shows that cruise missile engines present some unusual additional problems. Two prominent areas of design concern involve engine storage life and system survivability.

Improved storage life has a very strong influence on reducing missile engine system life cycle costs (LCC). The "wooden-round" LCC concept has been developed to focus attention on methods of improving environmental and handling impact on missile engines that normally spend almost their entire life in a dormant mode. Continuing work in this field is necessary to extend the storage life interval before maintenance must be performed. Attention must be focused on developing long-term environmentally stable lubricants, fuels, and pyrotechnic devices. Chemical treatment of parts to inhibit the growth of fungus and bacteria must also be addressed.

To develop the technology cost drivers, cost analyses for year 2000 gas turbine engine configurations, and the reference engine were evaluated by a life cycle cost approach. The life cycle cost was evaluated for each of the three phases of an engine acquisition: 1) development, 2) procurement (fly-away cost), and 3) operating and support. An in-house computer model for cruise missile engines was used in performing trade studies and determining the life cycle cost.

2.6.4. <u>Reliability</u>

Reliability not only determines the ability of the missile engine to satisfy the availability mission requirement, it has a direct bearing on the missile system cost.

Future technology trends will result in efforts to simplify basic engine configurations. Simplified designs promote low initial cost, improved reliability, and reduced cost of ownership. Trends leading to design simplicity will continue to be characterized by reducing the number of component stages along with the number of blades and vanes in an individual stage, developing substitute, lower cost materials, eliminating gearboxes by using direct drive accessories, and replacing complex hydromechanical controls with modular electronic units.

The challenge in reducing the number of component stages lies in developing efficient aerodynamic stages characterized by high pressure ratios, Mach numbers, and loading levels. Advancements in computational fluid mechanics are expected to be a major catalyst in this effort along with advanced experimental methods designed to define the strong secondary flow fields endemic to small aerodynamic components. The principal reliability variables traded during the evaluation process are design approach, design complexity, and structural/aerodynamic loadings.

2.6.5 Engine Concept Selection

Engine concepts formulated by these requirements will meet the goals of the individual technology drivers with varying degrees of success. The engine concept that best meets the requirements for the year 2000 cruise missile must be selected on the basis of clear superiority in the highest priority technology drivers. The order of priority is performance, size, cost, and reliability. Thus, if one engine provides the smallest size missile system but clearly falls below another engine in providing the required performance, the latter engine would be selected. However, if two engines result in essentially the same performance and missile size, cost and reliability then become the deciding factors.

3.0 TASK II - ENGINE CONFIGURATION AND CYCLE EVALUATION

Task II of the SECT program involved engine thermodynamic cycle and configuration evaluation. Engine concepts that offer the best mission/application payoff with reasonable risk for year 2000 applications were defined and evaluated.

Task II basically consisted of five steps:

Step 1 involved identifying engine design features and component technologies that offered the greatest opportunity for improvement, such as component efficiency gains.

Step 2 involved utilizing the SECT propulsive range mission to determine an engine design/operating point and size that was used in the parametric cycle analysis study.

Step 3 involved performing an engine design point parametric analysis to determine the best engine cycle obtainable to accomplish the year 2000 mission. In this step, a wide variety of engine parameters were investigated to optimize the engine thermodynamic cycle obtained.

In Step 4, the most promising engine cycles were selected and analyzed in greater detail. A review of common future engine technology requirements that led to the best available year 2000 engine were performed. The technologies selected for future study are those that will lead to maximum long-term payoff, and compatibility with long range goals and objectives.

Step 5 involved performing a part-load analysis of the selected year 2000 engine configurations, using existing steady-state simulation codes.

3.1 Engine Component Technologies

The first step in Task II involved identifying the engine design features that offer the greatest opportunity for overall system improvement. The primary technologies involved include improved aerodynamic design, improved component materials applications, improved manufacturing techniques, and advancements in recuperator technology.

Improvements in aerodynamic design, such as in the ability to accurately predict aerodynamic flow fields, can offer significant advances in engine component design and efficiency. This includes the ability to accurately model secondary flows and shock/boundary layer interactions.

Improved component materials applications constitute a technology that is currently being developed that can offer significant advantages, such as silicon carbide/silicon carbide (SiC/SiC) or carbon/silicon carbide (C/SiC) ceramic materials. Materials capable of handling higher turbine inlet temperatures, such as ceramics, can offer improvements in cycle efficiencies by reducing cooling flow losses and increasing allowable turbine operating temperatures. The lower mass of the composite/ceramic materials will result in lower inertia in rotating components, which will improve engine starting and transient response characteristics. Materials such as silicon nitride (SiN), or other ceramics, will be developed for use in high-temperature, high-effectiveness recuperators.

Improved manufacturing techniques that result in higher quality, more efficient aerodynamic components can offer improvements. Manufacturing methods such as automated precision fabrication and advanced assembly techniques will result in improved surface finishes, minimized tolerances, and lower endwall clearances, leading to improved component and cycle efficiencies.

Recuperator technology will be advanced by developing high temperature, high effectiveness units made of ceramic materials. These units may require new manufacturing techniques to achieve the desired structural configurations.

3.2 Engine Thrust Requirements

Selection of the optimum cycle for advanced technology engines depends on the maximum thrust required to provide the desired performance in the SECT reference airframe flying the reference mission. This thrust requirement, in turn, depends on the SFC characteristic of the advanced technology engine, because lower SFC will result in a smaller fuel load to accomplish the mission. This leads to a lower gross weight and, consequently, a lower thrust requirement for the same performance. To provide the needed guidance in the cycle selection process, it is necessary to estimate the relationship between SFC and the required net thrust to provide the desired performance. To do this, a preliminary propulsive range mission analysis was conducted using the reference engine by scaling the engine's SFC and subsequently its thrust to meet the desired performance. The required thrust was determined for the reference engine with normal SFC characteristics and SFC reduced by 10, 20, or 30 percent. This was accomplished using the Williams International-developed Integrated Systems Mission Analysis Program (ISMAP) under standard day conditions.

Mission/Performance Requirements

For the cycle analysis, a propulsive range approximation of the reference mission was used to determine fuel load required to accomplish the baseline reference mission. Published literature has indicated that operational range is on the order of 80 percent of the propulsive range. Therefore, a propulsive range of 3200 nautical miles (5926 km) was assumed to provide a reasonable representation of the baseline reference mission fuel requirement. The propulsive range mission profile is basically the same as the baseline reference mission with the exception that the 2000 nautical mile (3704 km) terrain-following segment is replaced by a 2600 nautical mile (4815 km) low altitude cruise segment as shown in Figure 7.

Mission Analysis

The propulsive range mission was first flown on ISMAP with the reference engine SFC characteristics (SFC scale factor = 1.0). Engine thrust was scaled up until the climb rate criterion at 600 nautical miles (1111 km) was met. Fuel weight was adjusted until the 3200 nautical miles (5926 km) propulsive range was achieved with virtually no fuel remaining, i.e. within ±1 lb (0.5 kg). As fuel weight was adjusted, the gross weight was also adjusted, as shown in paragraph 2.4, because of the change in missile length to house the fuel and the change in wing area to support the changed gross weight. The change in wing size also resulted in a change in the zero lift drag coefficient, as also defined in paragraph 2.4.

This change in drag combined with the change in gross weight consequently changed the climbing ability of the missile at the 600 nautical mile (1111 km) point in the mission. This necessitated a change in thrust scale factor in order to meet the climb rate criterion, and the whole procedure was repeated until it converged on the combination of fuel load and thrust scale factor that met the climb rate criterion and left virtually no fuel remaining at the end of the mission.

The optimum altitude for the first cruise portion of the mission was obtained iteratively by first establishing the configuration (fuel weight, gross weight, wing area) required to fly the 3200 nautical mile (5926 km) mission using an initial cruise altitude of 20,000 ft (6096 m). The change in residual fuel with change in initial cruise altitude while holding everything else constant was then obtained and plotted as in Figure 8. It showed the optimum altitude to be about 18,500 ft (5639 m). Negative residual fuel simply means that more fuel was required than was available in the configuration.

The minimum wing size was assumed to have maximum wing loading of 214 lb/ft^2 (10246 N/m^2).

Increasing wing size by five percent required more fuel than the minimum wing size as shown in Figure 8. This occurs because the increased drag due to lift for the smaller wing is more than offset by the reduction in zero lift drag. Also, the larger wing weighs more. This supports the wing loading assumption.

Results

The resulting thrust scale factor on the reference engine with its normal SFC characteristics was 1.938, which resulted in a maximum net thrust of 947.8 lb (4216 N) at sea level, Mach 0.7 conditions. The reference engine SFC at maximum power under those conditions is 1.07 lb/lb-hr (0.109 kg/N-hr). Improved SFC, however it might be obtained, would require less fuel to fly the 3200 nautical mile (5926 km) mission. This, in turn, would reduce the missile gross weight due to less fuel weight, less structure to hold the fuel, and less wing area to support the lower gross weight. The lower gross weight would, in turn, lead to a lower thrust requirement to meet the rate of climb criterion. The effect of improved SFC on the required thrust was obtained by running the same mission analysis as described above but with a scale factor on the reference engine SFC. Three levels of SFC improvement were studied using SFC scale factors of 0.9, 0.8, and 0.7, respectively. The results are presented in Figure 9. In addition to the normal fixed weight, a variation of fixed weight of ±100 lb (45.4 kg) was also studied to show the effect of engine weight relative to the reference engine. This effect is an important consideration when studying advanced technology engine concepts that are characteristically lighter or heavier than the reference engine. This data was used in conjunction with cycle parametric carpet plots of SFC versus specific thrust in order to determine what levels of the parameters would best satisfy the mission requirements.

3.3 Parametric Cycle Study

Component Efficiency

Size effects have a significant influence on engine component performance and the choice of the best cycle for a specified airframe application. The engine cycle parametric study was conducted so that this size influence could be quantified. This was accomplished by setting the polytropic and adiabatic efficiencies used in the cycle analysis code as a function of component weighted average flow parameter for axial compressors, centrifugal compressors, axial turbines, and radial inflow turbines. The compressor and turbine curves were then overlayed, and universal compressor and turbine curves developed that reflected the best possible efficiency attainable over a wide range of flow parameters. These universal curves imply that radial flow components are more likely to be required at very low values of flow parameter, while axial components will result in higher values of flow parameter.

The individual component efficiency predictions were developed using loss models imbedded in existing Williams International component preliminary design computer codes. These individual codes each address the separate effects of surface finish, tip clearance, and Reynolds number, in addition to a variety of loading and Mach-number-related parameters. In adopting this approach, there was an implied assumption that to achieve the quoted efficiency levels the component designers will select rational levels of loading and not be unduly restricted by geometric constraints.

Figures 10 and 11 present the universal size effects curves for three separate circumstances. The lower curves illustrate 1985 state of the art. The upper curves illustrate the ideal efficiencies attainable if all controllable physical impediments to losses are eliminated and only Reynolds number influences remain. The middle curves reflect Williams International's projection of component performance development progress attainable by the year 2000. These curves assume that 1) significant advances can be made in predicting aerodynamic flow fields more accurately (particularly secondary flows and shock/boundary layer interactions), and 2) production quality, in terms of minimizing tolerances, surface finishes, and endwall clearances, will improve. These improvements will be achieved through the increased use of advanced structural design analysis, innovative mechanical concepts, and automated precision fabrication and assembly efficiency.

Component efficiency levels were reviewed and based on the current efficiency levels of very large turbofan engines. The following table shows current component efficiency levels for these large engines and projected levels for the year 2000.

	EFFICIENCY	
	1985	2000
Fan (Polytropic)	0.90	0.91
Compressor (Polytropic)	0.91	0.92
Turbine (Adiabatic)	0.925	0.94

The efficiency scaler for a given component was determined in the following manner: first, the component inlet and outlet flow parameter were determined. From these two values, the delta flow parameter across the component is known. By calculating the flow parameter and efficiency across 10 equally spaced deltas between the inlet and output of the component, a series of 11 flow parameter/efficiency scalar pairs can be determined from the universal size effects curves. These 11 efficiency scalars can then be averaged, giving a flow parameter weighted average efficiency scalar that reflects the change in flowpath size across a given component. By applying the efficiency scalars obtained in this manner to the above efficiency levels, the effect of engine size on engine and component performance can be evaluated.

Advanced Turbofan Cycle Optimization

The first step in this evaluation is to perform a parametric cycle analysis using the above methods for calculating efficiency, and the following assumptions:

Advanced Turbofan Net Thrust = 780 lb (3469 N)
Sea Level, Mach = 0.7, Standard Day
Inlet Pressure Recovery = 0.944 (Flush Inlet)
Base efficiencies:

Fan: 0.91 (Polytropic)
Compressor: 0.92 (Polytropic)
Turbine: 0.94 (Adiabatic)
Burner: 0.998

Burner Pressure Loss: 3.5%
Mixing Plane Velocity Ratio: $Vs/Vp = V_{Bypass}/V_{Core} = 0.65$

Engine cycles were evaluated for the above assumptions at the following engine conditions:

Fan Pressure Ratio : 1.4, 1.7, 2.0, 2.3
Overall Pressure Ratio: 6, 8, 10, 12, 16, 20, 24, 28, 32
Turbine Inlet Temperature: 1600°F, 1800°F, 2000°F, 2200°F, 2400°F, 2600°F, 2800°F, 3000°F, 3200°F, (871°C, 982°C, 1093°C, 1204°C, 1316°C, 1427°C, 1538°C, 1649°C, 1760°C).

The thrust level of 780 lb (3469 N) was assumed based on Figure 9 and an initial estimate of a 20 percent SFC improvement.

The data from these cycle studies were carpet plotted, for a given fan pressure ratio, in the format of SFC versus specific thrust, with lines of constant TIT and overall pressure ratio (OPR), and are shown in Figures 12 through 15. Figure 16 shows a cycle study at a 1.7 fan pressure ratio, and was calculated without applying any efficiency losses due to size effects. A comparison of Figures 13 and 16 demonstrates the influence of engine component size on cycle performance. As TIT and overall pressure ratio increase, the engine's components become smaller and their efficiency decreases. Therefore, cycle SFC and specific thrust are higher when compared to a cycle that does not account for size effects.

An examination of Figures 12 through 15 indicates the effect of operating conditions on engine performance. The curves indicate that for a given TIT, engine SFC decreases with increasing overall pressure ratio. This rate of improvement decreases as OPR continues to increase. For a given OPR, engine SFC decreases as TIT increases, up to an optimum TIT, at which point engine SFC begins to increase with increasing TIT.

As a guide in selecting an appropriate cycle from these data, a rate of SFC improvement for a given increase in OPR and TIT was established. An SFC improvement of one percent for a 15 percent increase in OPR ((delta SFC/SFC)/(delta OPR/OPR) = (0.01/0.15)) was selected to provide good SFC levels while controlling the amount of turbomachinery required. An SFC improvement of 0.5 percent for a 100°F (56°C) increase in TIT ((delta SFC/SFC)/(delta TIT) = (0.005/100)) was selected to provide good SFC levels while controlling cost for improved turbine materials. By plotting these derivatives on each of the parametric cycle study carpet plots, the best engine cycle can be selected as the intersection of these two derivative lines. An example is shown for the fan pressure ratio of 1.7 plot in Figure 17. This results in four engine cycles, one for each fan pressure ratio, with an associated SFC, specific thrust, and bypass ratio for each cycle. Then, plotting SFC versus bypass ratio and fan pressure ratio for these four cycles (Figure 18), the curve shows the cycle at a 1.7 fan pressure ratio providing the best performance available while minimizing the cost and complexity of the required turbomachinery. The turbofan cycle selected has the following characteristics:

Fan Pressure Ratio:	1.7:1
Overall Pressure Ratio:	22.0:1
Turbine Inlet Temperature °F (°C)	2200 (1204)
Specific Thrust (FN/WA) lb•sec/lbm (N•sec/kg)	18.9 (185)
SFC: lbm/lb-hr (kg/N•hr)	0.7655 (0.07808)

A comparison of component efficiencies with the reference engine for this cycle is shown in the following table, demonstrating significant component efficiency improvements:

	REFERENCE ENGINE	SECT TURBOFAN	IMPROVEMENT
Fan Efficiency (Polytropic)	0.836	0.901	+0.065
Compressor Efficiency (Polytropic)	0.840	0.888	+0.048
Turbine Efficiency (Adiabatic)	0.872	0.906	+0.034

Recuperated Turbofan Cycle Optimization

The best cycle for the recuperated turbofan was selected in the same manner as for the nonrecuperated turbofan. The base assumptions were the same as for the nonrecuperated turbofan, with the following additional assumptions for the recuperator:

Recuperator Effectiveness = 0.65
Recuperator Hot Side Pressure Loss = 10 percent
Recuperator Cold Side Pressure Loss = 2 percent

The same cycle selection methods were applied as for the unrecuperated turbofan. The parametric plots are shown in Figures 19 through 22. The intersecting derivatives are shown in Figure 20 as an example but omitted from the remaining plots for clarity. The results of plotting the cycle selected by the intersecting derivatives is shown in Figure 23 and again shows that a fan pressure ratio of 1.7 provides best performance. It was decided, however, that an aggressive technology program would utilize a high-technology recuperator with a design effectiveness of 0.85. The cycle parametric study was repeated for a fan pressure ratio of 1.7, with a recuperator effectiveness of 0.85 (Figure 24). It was assumed that the recuperator effectiveness would not influence the fan pressure ratio at which the best performance cycle would optimize. The resulting best SFC recuperated turbofan cycles have the following characteristics:

	RECUPERATOR EFFECTIVENESS 0.65	0.85
Fan Pressure Ratio	1.7:1	1.7:1
Overall Pressure Ratio	10.5:1	8.0:1
Turbine Inlet Temperature: °F (°C)	2600 (1427)	2650 (1454)
Specific Thrust: lb•sec/lbm (N•sec/kg)	18.09 (177.4)	19.01 (186.4)
SFC: lbm/lb-hr (kg/N•hr)	0.7284 (0.0743)	0.6663 (0.0680)

The recuperated cycle with an effectiveness of 0.65 shows a 4.85 percent SFC improvement over the year 2000 advanced turbofan, while the cycle with a 0.85 effectiveness shows a 13.0 percent SFC improvement.

The use of a recuperator with a 0.85 effectiveness as opposed to 0.65 effectiveness further increases engine weight and volume. The higher effectiveness recuperator requires a switch from a cross-flow to a counterflow design. A counterflow design at the 0.85 effectiveness is at least double the volume of a 0.65 effectiveness recuperator. The improved SFC of the 0.85 effectiveness cycle is achieved, therefore, with a penalty of higher engine weight and volume. The fuel savings from improved SFC will have to be evaluated in view of the increased engine weight and size in terms of the overall results on missile size.

A comparison of component efficiencies with the reference engine for this recuperated turbofan cycle shows the following improvements:

	REFERENCE ENGINE	RECUP. TURBOFAN	IMPROVEMENT
Recuperator Effectiveness		0.65/0.85	0.65/0.85
Fan Efficiency (Polytropic)	0.836	0.901/0.901	+0.065/+0.065
Compressor Efficiency (Polytropic)	0.840	0.893/0.894	+0.053/+0.054
Turbine Efficiency (Adiabatic)	0.872	0.920/0.922	+0.048/+0.050

In comparing the efficiency improvements attained in the conventional turbofan and the recuperated turbofan cycles, a larger efficiency improvement was seen in the recuperated cycles. This can be attributed to the lower overall pressure ratios required in a recuperated cycle to obtain low SFC. These lower overall pressure ratios result in larger components and flowpaths, which hold size effects losses to a minimum.

As can be seen, the efficiency levels of the 0.85 effectiveness cycle are lower than the 0.65 effectiveness cycle, in spite of the lower overall pressure ratio cycle. In this case, the higher effectiveness recuperator improves the cycle thermodynamic efficiency, thereby reducing the quantity of airflow needed to supply the required thrust. This results in a corresponding decrease in flowpath size and component efficiency.

3.4 Year 2000 Advanced Turbofan Engine Description

The SECT year 2000 turbofan engines utilize the thermodynamic cycles optimized in the earlier work of Task II. The advanced turbofan engine is a 3.85 bypass ratio, twin spool engine, with an overall pressure ratio of 22:1 and a turbine inlet temperature of 2200°F (1204°C) (Figure 25). The cycle is unique in that it achieves a 22:1 pressure ratio utilizing only three stages of compression and 2-1/2 stages of expansion. This compares with current two-spool, 14:1 pressure ratio designs that require five stages of compression and three stages of expansion. The reduction in stages was made possible by using high pressure ratio per stage, moderate through-flow rates, and high tip speeds to control loading. The materials used in this engine are listed in Table IV, resulting in the weight breakdown shown in Table V. A summary of the aerodynamic component characteristics is provided in Table VI.

The 1.7:1 pressure ratio fan features a rotor and stator with low aspect ratios in order to reduce sensitivity to the distortion effects of the flush inlet. Possible use of a plastic fan will permit a broad range of untwist, as a function of engine speed, to optimize incidence and further increase fan stability at part power or cruise conditions.

The core axial compressor stage is driven off the high pressure shaft and achieves a 1.8:1 pressure ratio at maximum power conditions. This stage is a low aspect ratio design employing moderate through-flows, tip speeds, and loading levels. The efficiency and operating range goals of this compressor will be challenging, in view of the low aspect ratio nature of the design and the variable inlet flow field characteristics expected as fan pressure ratio varies with engine speed. Passive tip clearance control will be necessary on this stage, and will be critical due to the small component size.

The high pressure centrifugal compressor achieves a 7.19:1 pressure ratio. The rotor will be fabricated from lightweight, low inertia materials such as titanium-aluminide or composite ceramics. The rotor features a 45 degree (0.785 rad) backsweep, a

2400 ft/sec (732 m/s) tip speed, and a shroud that passively conforms to the rotor over a broad range of speeds and temperatures. A splitter-vaned, three-dimensional diffuser will be employed to achieve a compact engine center section. Surface finishes on this stage will be critical to achieving the desired performance goals.

A single 4.3:1 pressure ratio uncooled axial turbine will be used to power the high pressure rotor. This is a highly loaded stage with high levels of blade stress and tip speeds of 2300 ft/sec (701 m/s). A lightweight composite/ceramic material with high temperature capability will be used in this stage. The lightweight material will reduce rotor inertia, which will improve engine starting performance and response to transient thrust requirements. Passive tip clearance control is required, and shroud materials will be selected to match rotor kinetic and thermal growth rates, in order to minimize tip losses. The rotor coating in the tip region will be of a self-healing variety, designed to permit intermittent rotor/shroud contact to occur without oxidation damage.

The low pressure turbine is a 1-1/2-stage, highly loaded, 3:1 pressure ratio turbine. The rotor will be made from a one-piece, high temperature superalloy casting. The half-stage stators at the turbine exit permit high rotor loading by removing the swirl from the resultant exhaust gases.

Overall, the advanced turbofan engine utilizing the above components and technologies will achieve a thrust of 780 lb (3469 N) at sea level, 0.7 Mach number, with an SFC of 0.7655 lbm/lb·hr (0.07808 kg/N·hr).

3.5 Year 2000 Recuperated Turbofan Engine Description

The SECT year 2000 recuperated turbofan cycle is a single spool, 3.66 bypass ratio engine, with an overall pressure ratio of 8:1 and a turbine inlet temperature of 2650°F (1454°C) (Figure 26). The engine features a single stage geared fan, a single high pressure centrifugal rotor, a single stage axial turbine, and a 0.85 effectiveness counterflow heat exchanger. The materials used in this engine are listed on Table VII, resulting in the weight breakdown shown in Table VIII. A summary of the aerodynamic component characteristics is provided in Table IX.

The single-stage, 1.7:1 pressure ratio fan utilizes many of the features of the fan used in the simple turbofan cycle. The fan gearing ratio of 0.548 allows the fan to run at an optimum speed for best efficiency and performance. The centrifugal compressor is a 4.7:1 pressure ratio unit, featuring a 50 degree (0.873 rad) backsweep and double splitter vanes. It will be fabricated from either titanium materials or composite ceramic materials.

The single-stage, 3.6:1 pressure ratio axial turbine is highly loaded with a high tip speed. It will be fabricated from lightweight composite/ceramics capable of high temperatures and will employ many of the features used in the advanced turbofan high pressure turbine.

The unique aspect of this cycle is the 0.85 effectiveness recuperator. The high effectiveness of this recuperator forced a switch to a counterflow design, as opposed to a cross-flow design, in order to hold down the volume of the recuperator. The recuperator unit will be fabricated of silicon nitride or other ceramic materials capable of handling the high temperature (2200°F (1204°C)) turbine exhaust gases. The recuperator will be placed in line, behind the gas generator, with the bypass duct enclosing the unit. Cold compressor discharge gases will enter the recuperator, traverse radially inward through the units heating passages, and then flow forward into the engine to enter the burner. After flowing through the burner and turbine, the hot exhaust gases will flow directly rearward into the recuperator, after which they will flow radially outward to merge with the bypass stream and exit the rear of the engine.

The mixing of the recuperator hot exhaust gases with the bypass stream will require the use of a variable mixing plane to achieve optimum part-load and off-design performance. This results from the increase in recuperator effectiveness as airflow through the recuperator changes at low operating speeds and high altitudes. This reduced airflow results in dramatically different exhaust gas temperatures and flow volumes entering the mixing plane. Use of a variable mixing plane will optimize the mixing process and achieve optimum thrust at the given operating conditions.

Overall, the recuperated turbofan cycle utilizing the above components and technologies will achieve a thrust of 780 lb (3469 N) at sea level, 0.7 Mach number, with an SFC of 0.6663 lbm/lb·hr (0.0680 kg/N·hr). An important factor that needs to be considered in evaluating this engine cycle is the tradeoff between the greatly improved SFC, the recuperator cycle supplies, and the increase in engine volume and weight caused by the recuperator. The recuperated turbofan described here is 31 inches (79 cm) longer and 231 pounds (105 kg) heavier than the advanced turbofan.

3.6 Summary of Critical Technologies

A review of the effects of component size on engine performance, cycle selection, and resulting engine configurations points to a few select technologies requiring focused effort. These include: high efficiency compression systems, leading to higher overall pressure ratio, lower SFC cycles; and the application of composite ceramics in turbines to allow higher TITs.

The effect of flowpath size on component efficiency has been described in detail in paragraph 3.3. By concentrating on improving the efficiency of components, especially those compression system components located near the high pressure end of a compression system, engine SFC can be lowered. These efficiencies can be improved by developing novel blading arrangements in high pressure, centrifugal compressors; increasing our understanding of three-dimensional aerodynamic flow fields; and improving manufacturing quality, especially in the area of improved surface finishes and reduced tip clearances. Most research in this area would be concentrated on the compressor rotor, where the potential for loss is greatest.

Employing turbines of composite ceramic material that can withstand high temperature flows without cooling will further improve SFC by eliminating losses associated with the use of turbine cooling air and by allowing higher TITs. Also, the lower weight of composite ceramics will lower the inertia of rotating components such as the turbine rotor and result in better engine starting and transient response performance.

4.0 TASK III - SYSTEM PERFORMANCE EVALUATION

Mission analysis was conducted to determine the benefits to be gained with the advanced technology cruise missile engine concepts selected in Task II relative to the reference cruise missile engine. Performance evaluation was conducted using the reference airframe defined in paragraph 2.4, sized for each engine to accomplish the reference mission defined in paragraph 2.3. Two advanced technology engines were evaluated; an advanced technology turbofan engine and an advanced technology recuperated engine. They were evaluated in terms of:

- Missile size and weight required to fly the reference 2600 nautical mile (4815 km) mission, and
- Mission performance including the range achieved in the alternate missions.

The engines were sized to provide similar rate of climb performance at initiation of terrain following. Size was restricted to that which would fit into the 20-inch (51-cm) inner diameter airframe.

In addition, the effect of using a slurry fuel rather than JP-10 on missile size and weight required to fly the reference 2600 nautical mile (4815 km) mission was evaluated.

4.1 Year 2000 Mission

The reference 2600 nautical mile (4815 km) mission was used to size the missile for each engine so that just enough JP-10 fuel would be available to accomplish the mission on a standard day. The reference mission and the alternate mission profiles (used to evaluate mission flexibility) were previously shown in Figure 1.

An existing terrain model was used in the mission simulations. For the high altitude approach missions, the approach altitude was optimized to result in the lowest fuel consumption for the mission. The resulting approach altitudes are listed in Table X.

4.2 Year 2000 Engines

The engines were sized on the basis of the Task II propulsive range study described in paragraph 3.2 and used JP-10 fuel. The criterion for sizing the missile engines was that the missile must have the same climb rate capability at the beginning of terrain following as the current SLCM has at the end of its mission. The propulsive range study assumed that a constant subsonic Mach cruise at low altitude would simulate the terrain-following segments if total range was increased to 3200 nautical miles (5926 km). This proved to be a rather conservative assumption partly because the average terrain-following altitude was considerably higher and resulted in 571 lb (259 kg) of excess fuel for the missile with the reference engine depicted as Case 1 in Table XI. When this airframe was resized to eliminate the excess fuel, the climb rate capability increased for the same engine due to the reduction in gross weight as shown in Case 2. By resizing the engine to the original requirement, the missile size and weight could be reduced still further (Case 3). Case 2 was selected as the reference engine/missile because it presented more reasonable guidance system requirements at a modest increase in missile size and weight.

Fuel consumption characteristics during the reference 2600 nautical mile (4815 km) mission and the alternate missions for all three engines is shown in Table XII. Terrain following accounts for 69 to 77 percent of the total fuel consumption while the approach uses 18 to 26 percent and all the dash phases together only account for five percent. Thus, the engine SFC during terrain following is of paramount importance in determining fuel consumption. During terrain following, the engine operates anywhere from flight idle to maximum power. Average operating points for the three terrain-following segments for all three engines are shown in Figure 27 together with the operating points at the start of terrain following. The operating points appear closer to the minimum SFC for the advanced engines. This is also evident in the optimum altitude cruise operating range shown in Figure 28. Here the operating SFC is five percent above the minimum for the reference engine, three percent above the minimum for the advanced turbofan and less than one percent above the minimum for the recuperated engine. Thus, not only do the lower minimum SFCs provide for lower fuel consumption in the advanced technology engines, but their flatter SFC characteristics provide an additional benefit in placing the operating points closer to the minimum.

The three engines are characterized in part by the TIT, which is compared at three points in the reference mission on Table XIII.

4.3 Year 2000 Airframes

The airframes for all three engines are similar in that they all are restricted by a 20.0-inch (51-cm) missile inside diameter (20.38 inches (51.77 cm) outside) and utilize wing loading of 214 lb/ft^2 (10,246 N/m^2). Details of how the airframes were configured are provided in paragraph 2.4.

4.4 Mission Analysis Results

The missile system size and weight characteristics that meet the 2600-nautical mile (4815 km) mission requirements with the three engines is given in Table XIV. Comparison of the advanced technology engines to the reference engine is shown in Table XV. Both advanced technology engines reduced missile size and weight substantially. The recuperated engine clearly provides the best fuel consumption characteristics, but its greater bulk results in a slightly larger and heavier missile than the turbofan configuration. The difference is relatively small, however, and indicates that 2600 nautical miles (4815 km) may be near the crossover point between the turbofan and recuperated engines; i.e., the lower fuel/gross weight characteristic of the recuperated engine may well produce the smallest/lightest weight cruise missiles for ranges somewhat greater than 2600 nautical miles (4815 km). Table X shows that the recuperated engine provides a very small range advantage for the alternate low level approach missions that require 35 percent more fuel for the approach than the optimum altitude approaches. The difference in performance and resulting missile size when using the two advanced engines is small enough to consider the engines as equivalent in regard to these two technology drivers.

4.5 Effect of Using a Slurry Fuel

The effect of using SF-2 carbon slurry in place of JP-10 fuel was also investigated. While boron slurry characteristically possesses a greater volumetric heat release potential than SF-2, the risk in overcoming the current problems in effectively converting this potential in a cruise missile engine by the year 2000 eliminated this fuel from consideration. Aluminum slurry is not compatible with the recuperated engine since it would tend to plug up the recuperator. SF-2 was selected because it is being specified by the U.S. Air Force for current slurry programs. Although it is much more dense than JP-10 (77.8 lb/ft^3 (1246 kg/m^3 versus 58 lb/ft^3 (929 kg/m^3)), it has a much lower heating value (LHV = 16,300 Btu/lb (37,914 kJ/kg)) compared to JP-10 (18,100 Btu/lb (42,100 kJ/kg)). Also, with current technology, combustor efficiency is expected to be no better than 0.975 compared to 0.998 currently experienced with conventional fuel. An advanced technology combustor (design and material) is expected to bring combustor efficiency up to 0.995 with smoke levels of current engines using conventional fuel.

Mission analysis determined that the range of the missile with the reference engine can be extended from the reference 2600 nautical miles (4815 km) to 2775 nautical miles (5139 km) with the same fuel volume by using SF-2. Although the missile length remains the same, fuel weight increased by 34 percent because of its greater density. This increases gross weight and, likewise, wing area by 21 percent (due to the constant wing loading criterion). Consequently, engine thrust must be scaled up by 4.3 percent to maintain the same specific excess thrust criterion. The increase in fuel weight does not translate directly into a proportional increase in range due to the reduction in LHV to 16,300 Btu/lb (37,914 kJ/kg), reduction of combustor efficiency to 0.975, and increased drag due to the larger wing. Thus, the 34 percent increase in fuel weight only increases range by 6.7 percent with current state-of-the-art technology. It is important to recognize that these results are valid only with the above assumptions; i.e., it represents the improvement in range due to slurry fuel only, with no other changes to engine or airframe characteristics, except that combustor efficiency is degraded and wing size increased.

Mission analyses were also conducted with the reference engine and advanced technology engines to determine the effect of using SF-2 slurry in reducing the missile size and weight required to accomplish the 2600-nautical mile (4815 km) reference mission. The advanced technology engines in this case incorporate the advanced technology combustors for burning SF-2 slurry. The results are shown in Table XVI. With slurry fuel, the fuel weight and gross weight increases (as well as wing area increase) are less pronounced with improving SFC, and missile length is reduced seven to eight percent. The advanced technology engines using slurry can reduce missile length by eight to nine feet (2.4 to 2.7 m) compared to the reference engine using JP-10.

A summary of the additional benefits of using SF-2 slurry fuel from two viewpoints is presented in Table XVII. The first is the reduction in overall missile length that is achieved for a fixed mission range (as discussed above). The second fixes the missile length to that required for JP-10 fuel, and looks at the benefit of using SF-2 in regards to increased range. The effect of the improved SFC for the advanced engines is evident here. The range improvement with either of the advanced technology engines is approximately ten percent compared to seven percent for the reference engine.

4.6 Engine Life Cycle Cost (LCC)

The reference engine LCC breakdown is presented in Table XVIII and is compared to that of the advanced turbofan and the advanced recuperated engine. Adjustment of costs was made to reflect material cost differences and reduced labor cost due to simplicity. Items that have made the engine simpler from the labor and material stand point are shown in Table XIX.

The impact of these changes on production unit cost as a percentage of the total cost of the reference engine is shown in Table XX. This table indicates that the advanced turbofan engine cost should be 33.3 percent of the reference engine. The reduction in production unit cost was somewhat less in the advanced recuperated engine (45.1 percent of the reference engine) due to the additional cost of the fan reduction gearing and recuperator.

In the operating and support cost categories, significant cost reductions are postulated for both the advanced turbofan and advanced recuperated engines over the reference engine. These reductions are attributed to:

- Need for only one major recertification during 20-year life cycle,
- Fuel burn improvement, and
- Simplicity of design with resulting labor and material savings.

Although it is not currently feasible with today's state-of-the-art engine, the LCC of the reference engine with only one major recertification is also shown on Table XVIII for the purpose of showing the impact of simply reducing the number of recertifications.

The fuel cost was a direct ratio of fuel burn from the reference engine to the expected fuel burn of each of the advanced technology engines. The cost of fuel is considered to be the same. The increase in storage time is due to:

- Increase in the storage life of pyro devices by the year 2000,
- Removal of oil-wetted lubrication system,
- Removal of seals that degrade with time, and
- The storage of engines in controlled nitrogen environments.

From this analysis the advanced turbofan engine would have the lowest total LCC.

4.7 Engine Selection

The mission analysis results show that missile performance and missile size are approximately the same whether the missile is propelled by the advanced recuperated engine or the advanced turbofan engine over the 2600-nautical mile (4815 km) mission. Even though the recuperated engine consumes 14 percent less fuel than the turbofan engine, the resulting reduction in fuel tank length is more than offset by the increased engine length. The selection of one engine over the other for the year 2000 cruise missile, therefore, cannot be based on significant performance or size differences. The determining criterion, then, becomes life cycle cost.

The estimates for LCC show that the advanced turbofan engine is 44 percent of the LCC of today's turbofan engine state-of-the-art (reference engine). The LCC of the recuperated engine is 53 percent of today's engine. The advanced turbofan engine is 17 percent lower in LCC than the recuperated engine, and therefore the advanced turbofan engine was chosen for the year 2000 missile. The technology planning required for Task IV was therefore based on the technology readiness needed for the advanced turbofan engine.

5.0 TASK IV - SMALL ENGINE COMPONENT TECHNOLOGY PLAN

A technology plan covering a broad spectrum of disciplines has been developed on the basis of the results of the study of advanced subsonic strategic missile engine and missions conducted as Tasks I through III of this program.

The objective of the plan is to define specific technology disciplines and component programs within these disciplines where significant engine and missile system payoff can be achieved by the year 2000. The definition of payoff, for this study, has been limited to quantifiable and verifiable improvements in engine thermodynamic performance or reductions in engine cost. Each of the programs defined has been limited to design, fabrication, and verification test at the component level. System benefit will be established by analysis.

5.1 Identification and Ranking of Technologies

Four technology programs have been identified as meeting the selection criteria. These programs and their principal objectives and payoff are presented in ranked order in Table XXI, with the program eliciting the greatest combined need and payoff first. The ranking process could be somewhat subjective due to difficulties in determining an acceptable method of quantifying the benefit of performance versus cost improvements. In this case, however, ranking was done on the basis of the contribution of each technology to the total reduction in LCC, since LCC was the deciding criterion for selecting the advanced turbofan engine.

The Solid-Lubricated Bearing program contributes by far the greatest LCC payoff accounting for 45 percent of the total LCC reduction. The Advanced Small Component Aerodynamics program and the Ceramic Composite Materials program make similar contributions to LCC reduction with the former holding a slight edge. While the Slurry Fuel Technology program contributes no LCC reduction, it provides a significant contribution to missile range enhancement for a given engine.

Each proposed technology program is defined in this section in terms of major objectives, anticipated payoffs, design and program approach, major milestones, related and necessary supporting technologies, inherent risks, verification procedures, and projected time frame for accomplishment. In each technology discipline where the potential for significant improvement has been identified, a current state-of-the-art technology baseline has been defined. This baseline is the reference from which all projected benefits will be measured during the verification phase of the component technology program. This baseline reflects both Williams International's and perceived industry standards of accomplishment.

5.2 Solid-Lubricated Bearing Program

5.2.1 Background

The lubrication systems of small expendable turbine engines often represent a considerable fraction of the engines' initial cost. In the case of the reference engine, it represents approximately 40 percent of the total unit production cost and 33 percent of the operating and support cost. Furthermore, conventional liquid lubrication systems impose several additional engine design and operational constraints that not only add to the life cycle cost of the entire missile system but also limit its operational capabilities.

In particular, liquid lubrication systems are one of the principal obstacles toward achieving extended storage life. Current storage lives for tactical and strategic missiles are typically four years. At the end of this period, the missile is recalled for refurbishment at considerable expense. If the storage period could be extended to ten years and one refurbishment eliminated, a significant reduction in engine system life cycle cost could be realized.

The liquid lubrication system imposes additional design constraints that manifest themselves in reduced reliability, restricted flight attitude, increased weight, reduced survivability, and increased parasitic loss. A brief account of each of these limitations follows:

- Engine reliability is reduced due to the need for complex bearing cavity sealing arrangements and scavenge systems. The multitude of external oil lines and fittings also presents opportunities for leaks and handling damage. The lubrication pump and fuel/oil heat exchanger are complex subsystems and pose reliability problems in themselves. In addition, studies of engine survivability indicate that external oil lines and the fuel/oil heat exchanger have a significant impact on engine survivability.

- The requirement for liquid lubrication limits the operational and storage attitude of the engine. These constraints are mostly driven by venting and drainage system requirements.

- The complexity of the mechanical arrangement around seals, and the presence of a fuel/oil cooler, lubrication pump, oil lines, oil tank and the oil itself adds considerably to engine weight. These pieces constitute approximately ten percent of engine weight.

- The viscous drag due to engine lubricant is small compared to overall engine output power but contributes the performance equivalent of one point in engine specific fuel consumption.

The liquid lubrication system and its limitations could be eliminated by replacing the liquid lubricated bearings with ceramic dry lubricated bearings. As a means of implementing this goal, a program is outlined that addresses the problems attendant upon running ceramic hybrid solid lubricated bearings up to 800°F (427°C) and full ceramic bearings up to 1200°F (649°C). This work would have broad application in the unmanned expendable missile engine field and ultimately could benefit manned atmospheric as well as transatmospheric and space vehicles.

5.2.2 Program Summary

An advanced technology program is presented for moderate temperature (800°F (427°C)) hybrid metallic/ceramic rolling element bearings and high temperature (1200°F (649°C)) full ceramic rolling element bearings, with both concepts utilizing self-contained solid lubrication. The moderate temperature bearings will be constructed of metallic races, silicon nitride ceramic balls/rollers, and a metallic cage with solid lubrication imbedded in the cage. The high temperature bearings will be constructed of ceramic or cermet races, silicon nitride ceramic balls/rollers, and a solid metallic or reinforced ceramic cage with solid lubrication imbedded in the cage. The primary goals are to eliminate the oil lubrication and establish a design concept that will:

- Increase engine application operational domains due to increased temperature capability of the bearings in two categories:
  - Moderate temperature (800°F (427°C)) compressor-end or less hostile turbine-end support bearing positions, and
  - High temperature (1200°F (649°C)) hostile turbine-end support bearing positions.
- Reduce engine weight and volume due to elimination of the liquid lube system,
- Improve engine performance due to reduction in sealing requirements around the bearings,
- Result in unrestricted flight attitude,
- Increase long-term storageability and reduce maintainability in long-term storage concepts,
- Improve engine simplicity and survivability through elimination of liquid lube system components,
- Increase engine reliability due to the inherent system simplicities, and
- Eliminate low-temperature starting problems normally associated with high oil lubricant viscous drag.

This program encompasses basic material/tribological studies and the design, fabrication, and test of solid-lubricated rolling element bearings. Program efforts are complemented by state-of-the-art bearing analysis and bearing design computer code generation. The test program is structured to provide design/analysis correlation and verification leading to a technological design tool for use in the design of advanced engine bearing support systems.

5.2.3 Program Approach

The technical approach for this bearing concept is illustrated in Figures 29 and 30. It consists of a program in two phases. Phase I is concerned with verifying the solid-lubricated bearing concept and provides a stepping stone for Phase II. The technical information and data obtained from Phase I will be used as a data base for Phase II, which concentrates on the development of new technology for the concept.

5.2.3.1 Phase I - Concept Verification

Phase I consists of the design of a moderate temperature 800°F (427°C) solid-lubricated hybrid bearing, based on existing information and literature. The moderate temperature hybrid bearing will then be rig tested and engine tested in an existing test bed engine to validate the solid lubricated bearing concept. The tasks in Phase I, Concept Verification, shown in Figure 29 include:

- Selection of bearing materials/coating using available data,
- Selection and screening of existing solid lubricants using bench tests,
- Bench testing the bearing materials and lubricant materials to characterize their friction and wear properties,
- Bearing dynamics analysis implementing the bench test results to predict bearing performance,
- Design of a hybrid bearing for the number one position of the existing testbed engine with three alternate cage concepts,
- Fabrication of the hybrid bearings in each of the three cage concepts,
- Testing the bearings in a rig duplicating engine speeds, loads, and temperatures,

- Test and analysis verification/correlation using the rig test data, and

- Verification of bearing performance in a specially instrumented existing testbed engine, number one bearing position, at design speed conditions. The purpose of this test will be to verify the final hybrid bearing design by an endurance test of up to 10 hours. The engine will be operated in a stepped speed test procedure up to maximum engine design speed of 52,000 rpm. This will represent a bearing DN speed of 1.3×10^6. At each speed increment, the engine speed will be maintained until the bearing reaches stable temperature operation before proceeding to the next higher speed increment. Once reaching maximum speed, the bearing will be endurance tested up to 10 hours.

5.2.3.2 Phase II - Technology Development

Having verified an 800°F (427°C) self-lubricating hybrid ceramic bearing, Phase II would proceed with the technology development of this bearing concept. Using the information obtained in Phase I as a data base, Phase II would consist of the design and rig testing of both a moderate temperature, 800°F (427°C), solid-lubricated hybrid bearing and an enhanced high temperature, 1200°F (649°C), full-ceramic solid-lubricated bearing. Also in Phase II, computer codes to predict bearing performance and wear life will be generated and the manufacturing technology requirements for these bearings defined. The tasks in Phase II, Technology Development, shown in Figure 30, include:

- Development of bearing materials/coatings,

- Development of solid lubricants in the form of compacts, powders, and coatings,

- Continued bench testing of the bearing materials and lubricant materials to identify and characterize the optimum material combinations relative to their friction and wear properties,

- Bearing analysis and computer code generation to predict bearing performance and wear life,

- Design of a moderate temperature hybrid bearing, a high temperature full ceramic bearing, and the bearing mounting, cooling and powder delivery systems,

- Fabrication of the hybrid bearing, the ceramic bearing and the cooling and powder delivery systems,

- Conduct bearing rig tests for bearing/systems development and for verification tests under simulated engine operating conditions, and

- Test and analysis verification/correlation using the rig test data.

5.2.3.2.1 Bearing Materials/Coatings

The interactions between the rolling elements, bearing rings, and cage are an interactive process. Therefore, the development of materials and lubrication concepts for the various components cannot be conducted independent of each other. Materials development, lubricant development, and bearing development will be conducted in a concurrent and interactive process. Rolling/sliding high temperature traction testers will be used for material/lubricant identification, optimization and characterization, and for making the full-scale bearing test scheduling sequences more efficient.

The selection of effective bearing materials is a function of their thermal, physical and mechanical properties, as well as the operational environments and engineering constraints of the application. The available literature contains little quantitative information concerning the key properties and critical characteristics of materials required for application in high temperature solid-lubricated bearings. Consequently, an experience base of the interactions of materials as components of these bearing systems will need to be established.

Solid-lubricated bearings will generate a substantial amount of heat (due to high speeds and the fact that solid lubricants are poor heat transfer agents compared to liquid lubricants). In addition, these bearings must be integrated into an advanced turbine engine environment. Thus, thermal properties of materials are a primary consideration. The thermal properties that will need to be quantified are those that are basic to the appropriate bearing design and include thermal expansion, thermal conductivity, thermal diffusivity (the ratio of thermal conductivity to specific heat), thermal stress or shock resistance, and thermal fatigue.

The physical properties of greatest importance in high temperature bearings are dimensional stability, melting point, transformation temperatures and surface properties. Surface coatings must be selected that are compatible with the substrate materials in these regards. Specific mechanical properties are required of bearing materials at high temperature. Hot hardness is a primary criterion as it represents the ability of the material to maintain raceway surface hardness at the maximum operating temperature. High hardness, up to a certain point, is required to minimize asperity penetration and welding, and provide sufficient ability to support high contact loads. Other mechanical properties required of bearing materials for load-carrying ability at high temperature include rolling contact fatigue, tensile stress, stress rupture strength, modulus of elasticity, yield strength, creep strength, and Poisson's ratio.

Among chemical properties, oxidation resistance is of paramount importance for bearings and solid lubricants that operate in high temperature air.

At temperatures as high as 800 and 1200°F (427 and 649°C), the list of candidate materials that maintain strength and hardness while possessing rolling bearing performance qualities is somewhat diminished. Various super alloys, cermets and ceramics will need to be investigated. The prime candidate rolling element material, i.e., ball or roller material, for this program is silicon nitride. This material has been the preferred material for advanced technology ball and roller bearings, and has also been successfully employed for bearing races. This material is available as the Norton NC132 hot-pressed version or the Norton NBD100 hot isostatically pressed version.

5.2.3.2.2 Solid Lubricants

Solid lubricants are a key element in the development of moderate temperature 800°F (427°C) and high temperature 1200°F (649°C) bearings. Various forms of the solid lubricant and their means of replenishing the critical bearing rolling/sliding contacts must be established by proper development and materials selection. Initially the bearing critical surfaces must be coated with solid lubricants to permit the initial bearing run-in. The self-lubricating cage concepts must be supported by solid lubricant compacts, permitting good lubricant transfer, controlled wear rates, and good structural strength for high speed operation. Powder solid lubricant delivery systems will be included to aid in tailoring the bearing system and the solid lubricant compacts and also as a back-up system.

5.2.3.2.3 Friction and Wear Bench Tests

The primary purpose of bench test evaluations will be to provide a tribological assessment of material candidates and lubrication parameters to establish an integrated systems selection of materials and design. The assessment will include:

- Measurement of traction, both sliding and rolling (friction coefficients), which are necessary to model and predict heat generation in bearings, and

- Measurement of wear of lubricant coatings and bearing materials, which is necessary to select the most compatible material combinations, establish lubricant replenishment requirements, and estimate life of bearing components.

The data generated from tests will be used in the design of hybrid and all ceramic bearings. Specifically, the reduced data will be used in the formulation of the semi-empirical friction heat generation and wear rates computer models used to predict bearing performance.

In order to provide information that will be most useful in predicting and modeling bearing operation, laboratory test configurations and operations must simulate those found in the bearing as closely as possible. SKF Industries, Inc. has built and run an unique high temperature traction tester specially to study high temperature solid-lubricated rolling/sliding behavior under load, speed and temperature conditions representative of solid-lubricated bearings. This tester allows the direct measurement of traction forces as a function of several operating variables, as well as examination of the surfaces for tribological characterization. The resulting traction data will be used to develop analytical models to assess solid-lubricated bearing performance as a function of ball/raceway contact/lubrication conditions. These analytical models will form the basis for the design of solid-lubricated rolling element bearings.

The high temperature contact simulator was designed to measure traction force transmitted by a solid-lubricated contact between a rotating disk and a ball under controlled amounts of contact load, rolling and sliding, at temperatures up to 1300°F (704°C). Specimens are enclosed in a furnace with openings for specimen drive shafts and the lubricant burnishing mechanism.

5.2.3.2.4 <u>Bearing Analysis</u>

Engineering analysis will be performed to determine critical operating parameters and to establish the manufacturing requirements necessary for bearing fabrication. This analysis will characterize the bearing heat generation/cooling, dynamic, structural and wear characteristics. This data will then provide the basis for designing the bearing for optimum performance and for defining the necessary manufacturing controls to achieve the performance requirements. Rig tests will be conducted using the resulting bearing designs. The analysis will be correlated with the test data as the testing progresses and is completed. Sensitivity studies on analysis variables will be performed. The analysis will cover the following areas and will result in computer code being generated to predict bearing behavior:

- Bearing Heat Generation Analysis,
- Bearing Thermal System/Cooling Analysis,
- Bearing and Cage Dynamics Analysis,
- Cage Structural Analysis, and
- Bearing Component Wear Analysis.

5.2.3.2.5 <u>Bearing and Systems Design</u>

Engineering design studies will be performed to obtain optimum designs based on Phase I results for:

- A moderate temperature 800°F (427°C) hybrid ball and roller bearing for test rig development, and
- A high temperature 1200°F (649°C) ceramic ball and roller bearing for test rig development.

The moderate temperature hybrid ceramic bearings are high speed, solid-lubricated designs capable of operation up to 800°F (427°C), where the balls/rollers are made of silicon nitride, the inner and outer rings are made of a metallic material and the cage has self-lubricating material features at each critical sliding contact.

The high temperature ceramic bearings are high speed, solid-lubricated designs capable of operation up to 1200°F (649°C), where the balls/rollers and the inner/outer rings are made of silicon nitride and the cage has self-lubricating material features at each critical sliding contact.

These bearings will be designed as a cumulative result of the material/lubricant selections, the tribology bench tests, and the engineering analysis performed. The bearings will be capable of meeting the following design requirements:

- 800°F (427°C) for hybrids and 1200°F (649°C) for full ceramic bearings,
- 1.3 to 1.6 x 10^6 DN bearing speed (where D = bearing bore in millimeters and N = shaft speed in rpm),
- 100 to 200 lb (445 to 890 N) thrust load,
- 40 to 50 lb (178 to 222 N) radial load, and
- 10-hour bearing wear life.

The design of the bearings will include an optimization of the internal geometry parameters such as the raceway curvatures, the operating contact angles, and the cage ball pocket and guiding land clearances. The cage design will provide both structural strength and proper solid lubricant film transfer to the critical rolling/sliding contacts within the bearing.

Three support systems will be designed in addition to the bearings. These systems include:

- Bearing mount system,
- Bearing cooling system, and
- Powder delivery system.

Heat generated in solid-lubricated bearings must be dissipated by some means of external cooling in addition to possible conduction heat transfer to cooler engine regions. It is necessary to provide for bearing cooling to prevent excessively high or runaway temperatures in the bearings. A convective air impingement cooling system offers the most potential in reducing the temperatures within the bearings and is a most logical means of cooling in a gas turbine engine. Supplemented by the thermal system/cooling analysis already discussed, a cooling system capable of engine integration with minimal engine performance penalty will be designed and fabricated for use in the rig tests.

5.2.3.2.6 Hardware Fabrication

Hardware will be fabricated for both the hybrid and the full-ceramic bearing designs. Ceramic bearing ring preform development for the full-ceramic bearing ring components will occur in order to identify the basic manufacturing requirements for near-net shaped ceramic bearing rings.

An air impingement cooling system and a solid lubricant powder delivery system will be fabricated for use in development testing of the moderate and high temperature solid-lubricated bearings. The powder delivery system will be similar to that developed in the current U.S. Air Force-funded "High Temperature Solid-Lubricated Bearing Development" program at SKF Industries, Inc.

5.2.3.2.7 Bearing Tests

All of the hybrid and full-ceramic bearing designs will be performance and endurance tested through rig testing under conditions simulating advanced turbofan engine operation. The bearing test rig module will be designed and fabricated in Phase I but will be utilized in both phases of the program.

The objectives of the rig design and test plan will be to:

- Verify the operational capability of a solid-lubricated rolling element bearing in an actual turbine engine environment,

- Obtain bearing performance data to verify computer bearing design/simulation results towards the establishment of the design technology for a solid-lubricated engine bearing system, and

- Establish a baseline of bearing performance and endurance from which improvements derived from variations on material, bearing design, or cooling can be determined.

Shakedown tests will initially be conducted on the rig in Phase I. This initial portion of the plan will include tests to verify the performance of various features that are incorporated into the rig design. Among these features are the cooling air supply system, the thrust load system, and the slipring instrumentation system. Bearing operating parametric tests will be conducted to measure the heat generation and heat dissipation of the test bearing under conditions of speed, load, and cooling airflow rates/pressures/temperatures. Bearing endurance tests will be performed with the objective of verifying the wear life capability of the solid-lubricated bearing concepts.

5.2.3.2.8 Analysis and Application of Test Results

Analysis of these bearing parametric experiments will provide the necessary data for developing traction curves and establishing the friction coefficients that consider the effects of speed (sliding velocity) and temperature. The endurance portion of the test schedule will provide the empirical constants for evaluating wear. By means of these studies it will be possible to establish a baseline for comparison of the friction and wear for various bearing designs and lubricants.

5.3 Advanced Small Component Aerodynamics Program

5.3.1 Background

Small engine component aerodynamic performance has traditionally lagged and not been able to achieve the absolute levels of performance demonstrated by large engine components. This situation is the result of fewer resources being available for small component technology investigations and inherent physical problems associated with small aerodynamic components, which prevents direct transfer of large component technology lessons. Consequently, it is important that specifically tailored design and evaluation programs and techniques be applied if dramatic improvements in the performance of small aerodynamic components are to be realized.

There are three major areas where significant differences between large and small aerodynamic components exist. First of all, configuration characteristics of small engines are considerably different from large engines. For example, the following are characteristic of the selected small advanced turbofan engine:

- Centrifugal compressor,
- Low aspect ratio airfoils, and
- Curved interstage ducting from mating axial to radial stages.

Second, there are significant flow field dissimilarities between large and small aerodynamic components, such as:

- Ratio of boundary layer thickness to passage height,
- Intermingling of endwall secondary flows,
- Reduced Reynolds number,
- Turbulence intensity level, and
- Percent leakage through seals and clearance gaps.

Third, some geometric features are difficult to scale such as:

- Leading edge radii/blade thickness,
- Trailing edge radii/blade thickness,
- Surface finish,
- Tip clearances/passage height,
- Airfoil positional and surface tolerances/pitch or chord,
- Airfoil taper ratio and other blade dynamic restrictions,
- Airfoil fillet radius,
- Cooling air hole geometry, and
- Obtrusive measurement probes.

These special characteristics of small aerodynamic components have resulted in significant component and engine system performance penalties. Existing small engine cycles are currently using aerodynamic components whose efficiency is several points below similar, equally loaded, larger aerodynamic components. This was illustrated in Figures 10 and 11 for compressors and turbines, respectively. For fan jet engines, penalties of up to nine percent in SFC and ten percent in specific thrust result.

Furthermore, the poor efficiency potential of small aerodynamic components has prevented the use of thermodynamically attractive higher pressure ratio cycles because the advantages of the increased pressure ratio are more than offset by the declining efficiency of the smaller parts.

The latter argument, when extended, suggests that poor performarce potential has prevented the gas turbine engine concept from being applied to a broad range of small ground-based power plant and aviation engine requirements. This is in sharp contrast to the big engine side of the gas turbine engine field, where system power, volume, or materials limits have limited the growth of engine physical size.

Small gas turbine engines with significantly improved levels of fuel efficiency and specific thrust will require dramatic breakthroughs in small engine component efficiency. This can best be achieved by identifying, quantifying, and overcoming the principal loss-generating mechanisms currently understood to limit performance. The most conspicuous loss-generating mechanism is believed to be associated with secondary flow influences. Because of their small size, the inner spool aerodynamic components are particularly and unusually susceptible to these influences.

Dramatic improvements in the efficiency of small components can be achieved over a period of time if a comprehensive step-by-step procedure to identify, quantify, separate, and ultimately reduce or eliminate the loss-generating mechanisms is undertaken. This can be achieved by undertaking the improved small aerodynamic component program described herein. It will also become apparent that the test program described will provide an extremely useful empirical database for small components where none currently exists. Benchmark data will be available to validate computational fluid mechanics codes. These data will further the development of a range of nonobtrusive measuring devices that are particularly well suited for small component test environments.

5.3.2 Program Summary

An overall component aerodynamics plan that uses upscaled hardware in order to greatly assist measurement accuracy and manufacturing fidelity is presented herein based on the following methodology:

- Current small turbofan component with good performance will be purely upscaled to allow careful measurements by laser two-focus and miniature probes.

- Performance of the upscaled component will be obtained in a test rig running at the Reynolds number and percent running clearance of its engine counterpart.

- Stage entrance conditions (boundary layer thickness, turbulence level, etc.) will be investigated along with the performance of two geometrical variations.

- Data analysis will lead to the design, fabrication, and rig test of three additional experimental components.

- Data analysis and upgraded computer models will guide the design of a new stage in the original flowsize.

- Back-to-back rig tests with the original small stage will quantify the performance advances achieved in the program.

The objective of the advanced small component aerodynamics program is to identify, isolate, and quantify the major loss-producing aerodynamic and geometric mechanisms that are peculiar to small turbomachine components and then demonstrate methods of reducing or eliminating their influence. In the process of accomplishing this work, several additional benefits could be realized. They include:

- Developing a broad and general empirical database to be used with current state-of-the-art analytic design tools to further develop and demonstrate techniques of controlling or eliminating loss-generating secondary flow influences,

- Providing a critique of conventional small component loss models that are substantially based on results from large engine component design and test efforts,

- Providing detailed, accurate, empirical data of sufficient resolution and quality that it can serve as validation of advanced computational fluid mechanics design code predicted flow fields,

- Developing improved micro-miniature and nonobtrusive flow field measurement techniques,

- Providing state-of-the-art component test beds that would be available for future Government or industry research, and

- Demonstrating the capabilities of the most advanced real time data acquisition and analysis equipment.

5.3.3 Program Approach

The objectives of the advanced small component aerodynamics program will be accomplished by designing and fabricating an upscaled aerodynamic component that can be evaluated in a low density test environment where all basic similarity parameters are reproduced and all small scale secondary flows accurately duplicated. The scale-up of the selected test article shall be sufficient that all significant geometric effects can be selectively eliminated or controlled. It will also permit existing nonobtrusive test equipment, such as laser two-focus anemometers, to be adapted to the test rig environment and a new generation of micro-miniature fully immersed probes to be evaluated.

The test article shall be based on an existing design that has been thoroughly tested using conventional techniques. It shall be judged to deliver good to superior performance, for its size, using current standards. This test article shall become the baseline for the verification test performed at the conclusion of this program.

The test facility, in addition to being adapted for a low density test section, shall be capable of permitting rapid and selective variation and measurement of the following parameters:

- Test article basic geometry,
- Selected clearances,
- Selected endwall bleeds,
- Inlet turbulence level,
- Inlet flowfield properties and radial profiles,
- Inlet boundary layer thickness,
- Inlet Reynolds number, and
- Test article speed, flowrate, and backpressure.

5.3.3.1 Axial Compressor Performance Example

A typical program sequence is illustrated in Figure 31.

The program begins with the upscaling of the selected test article. For the purposes of illustration we will assume it is a previously designed and tested axial compressor that has demonstrated good performance in the core of a small, high pressure ratio turbofan engine. In the design phase of the program the axial compressor is scaled up to a point where the following geometric variables can be controlled to variations of less than four percent of their reference value:

- Rotor span,
- Rotor blade fillet,
- Rotor blade thickness,
- Rotor throat area,
- Running clearance,
- Surface finish,
- Rotor blade leading edge radius,
- Rotor blade trailing edge radius,
- Rotor chord, and
- Rotor stagger angle.

In parallel with the upscaling process, a review of current design practices will be conducted to determine the most significant features of small axial compressors that contribute to loss generation and secondary flow development. The results of this study will be used to identify specific features of the test article that could be varied with the object of achieving significant short term improvements in overall performance. On this basis, two variants of the basic test article will be designed for rig test evaluation. Typical design modifications could be variation of aspect ratio and introduction of blade tip end bend. The prime design and the two variants would then be released for fabrication while test rig and facility design modification commences.

The test rig design effort will be concentrated on the development of controls and quick change adapters to modulate inlet suppression, turbulence level, endwall boundary layer thickness, and radial gradients of temperature, pressure, and velocity. The test section will be designed with contoured windows to accommodate laser velocimeter or optical pyrometer surveys. Mounting pads and bosses will also be incorporated at this time for continuous tip clearance measurement probes, and subminiature traverse and fixed probe instruments.

Upon receipt of the test hardware a full inspection will be carried out to ensure that all geometric design parameters and features are within acceptable tolerance bands. When inspection is complete the test article and test rig will be assembled.

Testing will commence with the prime test article. First, overall component performance will be established to ensure a useful data base will be developed. This will be carried out with the test facility operating at normal levels of turbulence and other previously defined inlet flowfield conditions. If performance expectations are satisfied, blade element data will be gathered using the subminiature traversing and fixed element probes. The effects of the presence of these probes on local as well as overall performance will be ascertained. All fixed probes that are not built into the test article will be designed for rapid withdrawal from the flowpath. Most of the subminiature instrumentation will be capable of full mechanical traverse to resting positions outside the flowpath. When blade element data is complete, laser two-focus surveys will begin at selected operating conditions. The three-dimensional wrapping of the flow field will be complete enough that origin, development, and dissipation of all secondary flow will be described. Real time online graphical displays with supplemental hardcopy output will be necessary to ensure the completeness and veracity of the data.

The entire three-step test process will be repeated with selected inlet variables such as turbulence level, boundary layer thickness, or Reynolds number modified. As inlet conditions are changed, selected operating conditions will be monitored to determine whether there is significant overall performance impact to justify collecting blade element and three-dimensional flow field data. Inlet conditions will be varied to practical limits and operating conditions will cover the useful spectrum of axial compressor operation.

This procedure will then be repeated for the two variant designs.

Data analysis will be conducted in parallel with test efforts. This will be possible due to the application of real time on-line data acquisition and analysis systems. Data processing to final tabular and graphical standards will be necessary to ensure complete and accurate results are achieved. The timely review of these data will also be critical to determining whether modifications to the run program are necessary and justified.

The results of these tests will determine the influence of several key geometric and aerodynamic parameters on small axial compressor performance and provide incentive for modifications to the initial design. The most important conclusions reached are expected to result from the three-dimensional flow mapping and identification of the source, magnitude, and interaction of secondary flows. These flows are currently considered the principal obstacle to major advances in small component aerodynamic performance.

Based on these results, a design iteration will be conducted where two or possibly three modifications to the prime and/or variant compressor design will be accomplished. Table XXII shows how one geometric variable (fillet size) could be investigated. The object, once again, will be to minimize the secondary flow influences through selective geometric modifications to the airfoils of flowpath endwalls. The redesigned parts will once again go through the same process of fabrication, inspection, test, and data analysis. At this point, one or more of the modified compressor stages will have demonstrated significant performance improvement when compared to the baseline concept. The best performing upscaled concept will then be selected and scaled down to the full size of the originally selected axial compressor. This part will be fabricated, inspected, and subjected to a back-to-back test with the original full size axial compressor. The principal purpose of this test will be to verify the magnitude of the performance improvement.

The program will be completed by issuing a comprehensive final report summarizing program results. The report will be supplemented by several interim progress reports where the extensive data base generated will be presented. This data base is expected to be of great immediate use for other compressor designers and in the development and checkout of advanced, sophisticated, computational fluid mechanic codes.

Figure 32 presents a proposed schedule for this work.

5.3.3.2 Other Aerodynamic Components

The procedure described above for axial compressor performance will also be applied to fans, centrifugal compressors, and axial turbines. For example, geometric variables for a centrifugal compressor are given in Table XXIII.

5.4 Ceramic Composite Materials Program

5.4.1 Background

Development progress in the large engine field has traditionally been paced by temperature limitations of combustor and turbine section materials. As new high temperature alloys have been developed, engines with thermodynamically more attractive cycles have been designed to deliver significant improvements in SFC, specific thrust, and thrust-per-pound of engine weight. Many of these high temperature large engine benefits have been realized by the application of sophisticated cooling concepts in combination with the advanced high temperature materials applications. The use of often complex and costly cooling schemes in large engine applications can be justified on the basis of the relatively long life of these typically man-rated systems and the considerable fuel cost savings that can be realized during operation over this extended period. Small engines, on the other hand, and particularly those that are expendable (one mission) designs, have not been able to capitalize on these high temperature cooled concepts due to the inherent cost constraints they are subjected to. Furthermore, even in situations where cost is not of paramount importance the intricacy of the internal cooling passages is out of reach for even the most sophisticated fabrication processes.

The prospect of high temperature, uncooled combustor and turbine sections for small expendable turbine engines could become a reality with the successful development of monolithic and composite ceramic materials. The fabrication processes used in the production of ceramic composite components are particularly suited to the manufacture of small, integral structures. Potential for low cost is provided by elimination of intricate internal passways and the abundance of the basic materials typically used. Fabrication and inspection costs must be reduced from current levels where they reflect the small production quantities involved. Because of the risks involved in developing new materials, initial applications in short life nonmanrated applications make sense from a safety point of view. Once this technology is "proven out", transition to manrated long life larger engines will be feasible. The payoffs associated with the emerging technology are summarized as follows.

- Increased performance through increased uncooled temperature capability,
- Lower density,
  - Reduced weight and
  - Reduced shaft inertia, faster acceleration and deceleration,
- Tailorable stiffness, reduced blade dynamics problems,
- Potential for reduced cost due to reduced complexity, and
- No strategic materials.

In order to realize the potential payoffs associated with uncooled ceramic composite turbine components, a series of material characterization efforts must be undertaken. Currently, there are no domestic sources for ceramic composites that have demonstrated the capability to fabricate structural components. It is known, based on evaluations of the materials made in France by Société Européènne de Propulsion (SEP), that ceramic composite materials have the potential to revolutionize turbine engine design. SEP has demonstrated the capability to fabricate complex components with impressive strength levels. The material systems available from SEP that appear most promising are listed below.

- Nicalon silicon carbide (SiC) fibers in chemical vapor deposited (CVD) silicon carbide matrices and

- Graphite fibers in CVD silicon carbide matrices.

Currently, these materials are made by chemical vapor deposition (CVD) methods, which are costly and time consuming. The strength levels and maximum use temperatures for these materials are summarized as follows:

- For structural ceramic composites:
  - Demonstrated mechanical properties
    - In plane tensile strength — 70 ksi (48 kN/cm^2)
    - Across ply tensile strength — 4 ksi (2.8 kN/cm^2)
    - Interlaminar shear — 6 ksi (4.1 kN/cm^2)

  - Upper use temperature limits
    - SiC/SiC — 2000°F (1093°C)
    - C/SiC — 3000°F (1649°C)
    - C/SiC — *3800°F (2093°C)

*With coatings provides growth capability

The major challenges that need to be addressed prior to implementation of state-of-the-art ceramic composites in turbine engines include the following current limitations:

- 2000°F (1093°C) limited Nicalon SiC fiber due to:
  - Crystallization
  - Oxidation of impurities
  - Embrittlement

- Higher purity (more stable SiC fiber required),

- Slow and expensive vapor deposition densification process,

- Currently limited two-inch (five-cm) part thickness by CVD process, and

- Domestic technology is far behind foreign capabilities.

Also, additional evaluation is required in the following areas:

- Definition of stability of the composites under subcritical load,
- Fatigue properties,
- Oxidation resistance of C/SiC materials,
- Embrittlement characteristics, and
- Design methodologies.

5.4.2 Program Summary

The objectives of the ceramic composite materials program are to accelerate the development of composite ceramic design and fabrication methodologies and to provide a firm foundation for the establishment of domestic supplies of this material. The approach addresses the evaluation of component designs using existing materials in parallel with a material characterization study effort leading to radical improvements in fiber qualities and fabrication architecture.

Initial groundwork is being conducted by Williams International (and possibly other engine manufacturers) in which the feasibility of ceramic composite turbine components is being evaluated. In these efforts the French materials are being used to provide an assessment of the capabilities and limitations of the material systems. This type of work should be continued over the next five years for two reasons. First, in working with the French materials, it is possible to achieve an early understanding of the design methodologies which will be required. Many of the critical design problems associated with attachments, interfaces, and fiber architectures can be identified and solved during these feasibility study programs. A large cross section of limited life and man-rated applications should be evaluated in order to identify the future role ceramic composites can play in turbine engine development. Second, comprehensive analysis of the most advanced available materials can provide insight into manufacturing methods for domestic sources. This could drastically shorten the time required for U.S. sources to approach the foreign technology.

In parallel with the feasibility study efforts, a comprehensive material characterization effort must be initiated. Material properties must be made available for current ceramic composites for use in preliminary component design studies. The French composites should be a part of the characterization program to provide a benchmark from which to estimate future capabilities. In characterization programs, material fabricators should work closely with engine designers in order to understand the requirements of the applications. The turbine designer's inputs will be required in the specification of suitable fiber architectures, design of test specimens, and definition of test parameters based on engine operating environments.

If a domestic source of supply for advanced ceramic composite materials is to be established, it is necessary to initiate a comprehensive government-supported thrust in this area. This thrust should be aimed at near term duplication of the foreign technology as well as long term alternative approaches. A licensing agreement between SEP and a U.S. source would be one alternative that would result in the most rapid technology development. There is little chance of this happening without government support because of the relatively small immediate market for the materials and the requisite long fabrication times and high expenses. As suggested earlier, material developers must work closely with engine designers to ensure that optimum material systems for near term utilization receive the focus of attention.

Concurrent with the establishment of domestic material sources, some in-depth assessment programs should be undertaken. In these efforts, specific component requirements would be identified and fiber architectures would be proposed to suit their requirements. Detailed analytical design studies would be required to provide a means of measuring material capabilities. In parallel with component designs, analytical efforts should address the micromechanics of ceramic composite material systems. Design methodologies will require modification in order to deal with the observed behavior of these materials. Specialized structural tests would validate the design methods and help define component configurations that are feasible for future engines.

Current ceramic composite systems that utilize silicon carbide as a matrix material are limited to temperatures of approximately 3000°F (1649°C). Materials that utilize NICALON silicon carbide fibers have a much lower temperature limitation. These limitations should be addressed in programs aimed at fiber improvements and incorporation of high temperature coatings. Efforts along the line of the current DARPA-funded Dow-Corning fiber development program should be continued and expanded upon. High temperature coatings, particularly for graphite fiber/silicon carbide matrix materials, should be investigated in order to increase the temperature limitations to 3500°F (1927°C) and above. Coatings currently being developed by Vought and United Technologies Research Center for ELITE applications may be suitable for ceramic composites as well.

In the 1990's it will be necessary to initiate manufacturing technology programs to deal with the complexities of composite component fabrication. These efforts will have to address the high cost and long lead times associated with today's manufacturing methods, while providing production quantity rate capability. Similar programs are now addressing these requirements for carbon/carbon components. The problems are expected to be even more challenging, however, because of the difficulty inherent in machining ceramic composites. Quality control and nondestructive evaluation techniques will require demonstration to deal with the unique characteristics to ceramic composite structures. Specialized test rigs will also be required for validation of design concepts at extremely high temperatures.

5.4.3 Program Approach

The proposed overall approach to the problems discussed above is summarized as follows:

- Evaluate current materials in in-depth manner,
- Define required characteristics for the required applications,
- Identify areas of deficiency,
- Bring domestic technology up to speed via studies of existing materials,
- Design engine components for durability tests and fabrication feasibility studies,
- Validate design methodologies through component and specimen tests, and
- Implement ceramic composites in small, limited life non-man rated engines for early assessment.

5.4.3.1 Domestic Readiness

The task labeled "U.S. Source Readiness" on the accompanying schedule (Figure 33) should begin as soon as possible. The objective of this task is to provide a domestic source for state-of-the-art ceramic composite materials similar to those currently available from SEP in France. The individual efforts required to accomplish this goal include the following:

Evaluation of the Current Technology

A detailed analysis of the currently available ceramic composite material systems is required to define the most promising processing methods. Evaluation techniques will include mechanical property tests, destructive analysis via scanning electron microscopy (SEM), metallography, x-ray diffraction, and similar methods. This effort will be a team effort involving a potential ceramic composite fabricator, a test and analysis source experienced in composite material techniques, and an end user capable of defining material property goals and application environments.

Process Identification

Based on the study described above, one or more processing approaches will be outlined for detailed evaluation. Different approaches may be required for the various alumina, graphite, and silicon carbide fibers that are of interest. Ideally, this effort will dovetail with the ongoing DARPA-funded fiber development program. Processing methods will be defined by end user property requirements.

Process Validation

The optimum processes identified in the above described task will be utilized to produce sample quantities of each type of ceramic composite material. The materials will then be subjected to the same evaluation techniques described earlier. The results of the test will be compared to those obtained from the state-of-the-art foreign materials to guide the development process.

Process Optimization

The processes will be optimized to reduce cost and lead times as well as to improve properties.

This will become an iterative program leading to a production material in the early 1990's. Figure 34 depicts the proposed program in schematic form.

5.4.3.2 In Depth Study

The following effort descriptions outline the approach for an in-depth component evaluation and feasibility assessment task as depicted in Figure 35.

Component Identification

In this effort, the design of several advanced engine concepts that could benefit significantly from implementation of ceramic composite materials will be reviewed. Based on these engine designs, several components that are potential applications for ceramic composites will be selected. Candidate components include combustor shells, nozzle structures, transition ducts, shrouds, seals, and rotors. A trade study will identify the components that provide the greatest potential payoffs.

Configuration and Architecture Selection

The components selected earlier will be studied to isolate the aspects of the designs that could pose fabrication challenges, or are critical from a structural standpoint. Generic components will be designed to embody these critical fabrication and structural aspects. The generalized components will be representative of the selected advanced engine components and will provide an assessment of the potential of the material fabricators to produce the required shapes with the required properties. The current fiber architecture capabilities of the material fabricator will be reviewed and several weave and layup approaches that show potential for gas turbine component applications will be selected. Architecture concepts will be identified from among those that the contractor has demonstrated capability to produce.

Test Plan

A test and evaluation plan for non-destructive evaluation (NDE) and destructive testing of the ceramic composite materials will be prepared. The test plan will detail the type of tests to be conducted, the quantities, temperatures, stresses, durations, and other pertinent parameters as well as anticipated results and objectives. The following basic property tests will be included in the test plan as a minimum:

- Tensile in-plane,
- Tensile across ply,
- Interlaminar shear,
- Compression,
- Stress rupture,
- High cycle fatigue,
- Thermal expansion, and
- Modulus of rupture.

Additional tests will be planned to evaluate the effects of part configuration on properties. The subelement specimens required for determination of the effect of complex configuration and size on properties will be designed in this task. Analytical designs will be conducted using a finite element code specifically developed to deal with nonlinear, composite material behavior. The specimen design analyses will ensure that the test articles will fail in the desired modes and locations. For the fabricability assessment, simulated components will be designed that represent the geometries of the engine hardware.

Specimen Fabrication

The mechanical property test material and the fabricability assessment specimens will be subjected to nondestructive evaluation per the test plan. Panels and parts will be inspected by x-ray, microfocus x-ray, laser holography, and ultrasonic techniques to identify internal defects or low density areas that may be present. Computer-aided tomography (CAT) will be evaluated as a potential test method and, if feasible, CAT tests will be conducted on the specimens. Records of the NDE results will be kept for later correlation with destructive test results.

The fabrication feasibility specimens will be inspected to determine the degree of dimensional fidelity achieved. Tolerances on the hardware will be representative of those required for serviceable engine hardware. The ability of the fabricator to maintain the tolerances will be established.

Destructive metallographic and SEM evaluations will be conducted on the flat panel materials as well as the simulated engine hardware. Data from the destructive evaluations will be used to determine the effect of microstructure and fiber/matrix interaction on mechanical properties. Comparison of metallurgical cross sections and fractographs between the flat panels and the simulated hardware will help identify the effects of configuration and thickness on properties and behavior. Attempts will be made to correlate defects with the results of the NDE tests conducted earlier.

The basic mechanical properties of the various fiber architectures will be determined via testing per the detailed test plan. Tensile, compressive and shear properties will be measured over a range of temperatures to establish the effect of temperature on strength. Oxidation resistance and strength retention after high-temperature soak will also be evaluated. Laser holography vibration analyses and flat disk spin tests will be conducted. The effect of part geometry on properties will be established by testing specimens cut from various locations in the simulated components and comparing the results. Complete failure analyses will be conducted on the test specimens to establish that undesirable failure modes are not induced.

The test data generated in this task will be reduced and analyzed. Properties of the various fiber architectures will be compared to properties predicted in analyses. The measured properties will also be compared to the known requirements of each ccmponent application. Areas where properties are inadequate will be identified.

Component Design and Architecture Selection

Using the measured mechanical properties, preliminary design studies will be conducted for selected advanced gas turbine engine hot section components. The analytical designs will demonstrate the feasibility of ceramic composite turbine components using state-of-the-art materials. The configurations of the components will be modified to suit the demonstrated capabilities and limitations of the tested materials. Based on the measured mechanical properties of the existing fiber architectures, a trade-off study to identify optimum fiber architecture concepts for the selected component applications will be conducted. These concepts will serve as targets for domestic suppliers. Recommendations for additional ceramic components study efforts will be prepared. The evaluations of these current materials will provide insight into design, fabrication capabilities, and testing of ceramic composite components that will be useful in dealing with future domestic materials.

5.4.3.3 Architecture Properties Improvement

Further optimization of fiber architecture and property characterizations will be conducted under a follow-on task as shown on the schedule and in Figure 36. The following efforts comprise the bulk of this task.

Definition of Design Requirements

During this task, the operating environments and projected design requirements of several potential ceramic composite component applications will be established. Potential applications include rotating and static uncooled engine structures. Some preliminary component configurations will be designed based on aerodynamic performance and structural requirements.

Architecture Definition

Rather than selecting the optimum available fiber architectures for these components, the purpose of this effort is to define new weaves and ply layups that particularly suit the geometry and stress requirements of each component. Properties of the new architectures will be predicted based on the results of mechanical property tests of the precursor building blocks. Input from a material supplier will be required to ensure that feasible architectures are defined.

Material Fabrication

The selected material supplier will fabricate samples of the new architectures and ply layups for evaluation. Nondestructive evaluation techniques will be used to assess initial quality of the material.

Material Evaluation and Design Validation

A series of mechanical property tests will be conducted to validate the analytical predictions of material properties. Based on the test data, component designs will be completed for hardware fabrication.

5.4.3.4 Manufacturing Technology

Manufacturing technology effort will be undertaken to optimize producibility of ceramic composite components for turbine engines. Cost and lead time concerns will be addressed via bulk processing and automation techniques similar to those being considered currently for carbon/carbon composites and graphite epoxy materials.

5.4.3.5 Coatings

Another task will be required in the 1990's to evaluate high temperature coatings for ceramic composites. This technology would allow operation above the current 3000°F (1649°C) limit. Definition of the efforts required in this task will depend on the results of current similar efforts in the carbon/carbon field. The manufacturing technology and coating programs are delineated on the schedule (Figure 33) to establish the projected time frames.

5.5 Slurry Fuel Technology Program

Slurry fuel technology advancements hold the promise for significant improvements in subsonic, strategic, missile range and, consequently, system effectiveness. The mission analysis in Section 3 showed that SF-2 (a 56 percent loaded carbon slurry fuel) could improve system range by ten percent over current capabilities.

5.5.1 Background

In 1978 Williams International began its work with slurry fuels by building a combustor test rig and conducting combustion experiments with fuels donated by the fuel manufacturers. Many potential problems were encountered, and novel solutions were required to solve some of the problems. For instance, the use of metering valves was found to be unsatisfactory for slurry fuels, and a fuel metering system based on a positive displacement pump was developed and utilized. Fuel injectors suitable for liquid fuels plugged when used with slurries. A double conical rotating slinger fuel injector was developed for use with slurry fuels.

In 1980 Williams International received support from several pump manufacturers who conducted tests for Williams International to determine how various pump designs would operate on slurry fuels. This work led to the decision to use positive displacement gear pumps to perform the pumping and metering function. The pump speed is controlled to set the flow rate to the combustor. Also, in 1980 Williams International conducted the first engine test with slurry fuel. This test demonstrated potential feasibility for use of slurry fuels in low residence time combustors. In 1981, Williams International was subcontracted by Sun Tech to perform screening tests on a variety of carbon slurry fuel formulations that led to the development of SF-2. Pumping testing at temperatures down to -65°F (-54°C) and combustion testing in both can-type and annular combustors was conducted. During these tests, combustion efficiencies up to 92 percent were achieved in a relatively stock cruise missile combustor.

In 1983 Williams International was subcontracted by Boeing Aerospace Co. to support a fuel system development program for slurry fuels funded by AFWAL. During this program, Williams International conducted engine cycle studies for three cruise missile missions (i.e., subsonic, supersonic, and subsonic with supersonic dash). Computer models and engine layout drawings were developed for all three cycles. From the results of this phase of the program Boeing selected Williams International to continue on to the hardware demonstration portion of the program.

In 1985 Williams International prepared for a new U.S. Air Force-funded program on aluminum slurry fuels by conducting combustion tests in a new high temperature combustor. This combustor used an oxide dispersion strengthened (ODS) alloy liner with a cast zirconia nozzle section. Five different aluminum slurry fuels were burned with varied success. During the last test, the flame temperature exceeded the capability of the liner material and caused the liner to fail. In future tests, liners made of columbium will be used.

5.5.2 Current Programs

Williams International is currently involved in four slurry fuel programs.

The Boeing Carbon Slurry Fuel System Program is entering the breadboard fabrication stage. Williams International will provide a fuel injection system simulator for use in the breadboard demonstration test. This device will simulate the backpressure on the fuel delivery system caused by the fuel injector and combustor. The rig will include a hydraulically driven slinger fixture and a pressure vessel simulating the combustor pressure.

Williams International will be a subcontractor of the Sun Refining and Marketing Co. Aluminum Slurry Fuel Formulation Program. On this program, Williams International will perform combustion tests on a variety of aluminum and hydrid carbon/aluminum slurry fuel formulations. Two types of combustors will be used. A can-type with air blast atomization will be used for initial screening tests. A slinger-type annular combustor will be used for the final optimization tests.

Williams is the prime contractor on an exploratory development program to develop solid hydrocarbon slurry fuels. The novel concept involved in the formulation of these fuels is to microencapsulate the soluble solid hydrocarbon resin in an insoluble phenolic material to maintain the two-phase slurry formulation. The physical properties of the slurry fuel are better at low temperature than a solution formed by dissolving the solid in the liquid carrier at low temperature.

Finally, Williams International is working on a combustor development program that is being jointly sponsored by AFWAL/NASA. This program extends the application of a current AFWAL program on high temperature carbon/composite combustor liners to a combustor designed for high efficiency combustion of carbon slurry fuel. The combustor will use swirl and stoichiometry staging to achieve combustion efficiency over 99 percent. The combustor will be tested in a high pressure rig at the NASA Lewis Research Center.

5.5.3 Program Summary

The objective of Williams International's program is to evolve a system that will operate efficiently on carbon slurry fuel and achieve the full increase in range that the high density fuel can produce. Figure 37 presents the program schedule and plan.

The program consists of three exploratory programs.

5.5.3.1 Combustor Aero Program

The combustor aerothermodynamic design, fabrication, and test program is a continuation of an existing program to optimize the aerothermodynamic design of a carbon slurry combustor fabricated with carbon/carbon liner wall. New tooling for the carbon/carbon parts will be designed and fabricated to obtain parts that give the best flowpath geometry.

5.5.3.2 Fuel Metering Program

The fuel metering program is a currently missing building block in the overall program plan to develop a carbon slurry engine. The program will design, fabricate, and test an electronic fuel control for operation on slurry fuel. The control will be based on a microprocessor to interpret sensor signals and provide command signals, and a hydraulic-driven positive displacement pump that will be used to do the actual fuel metering. The valveless positive displacement metering pump system has been demonstrated with liquid fuels on the cruise missile engine and has proven to be an effective means to control the fuel flow rate. The proposed program would link the digital microprocessor to the metering pump system to achieve a fuel metering system capable of operating on slurry or other high-viscosity fuels.

5.5.3.3 Integrated Engine Test Program

The integrated engine test program is a technology integration program in which the combustor and fuel system will be run together in an engine test bed. This program will provide a full scale proof of concept. The engine will be run in an altitude chamber to verify the capability to operate at environmental extremes. The system could be expanded to include the vehicle fuel system as well.

6.0 DETAILED SUMMARY OF RESULTS

Williams International has conducted studies to identify component technology requirements for substantial performance and cost improvements applicable to subsonic, strategic, cruise missile engines for the year 2000. The program was conducted in four separate phases and was concluded with a comprehensive technology plan.

In the first phase of the program, future requirements were defined and major contributors to mission success in the year 2000 were identified. These were distilled into five major mission success criteria:

- Avoid detection,

- Avoid defenses,
- Confuse defenses,
- Expand number of targets vulnerable to attack, and
- Protect the launch platform.

These place numerous and, in some cases, conflicting demands on a cruise missile system. The resulting cruise missile characteristics, in turn, impose multi-faceted demands on the engine. To facilitate the evaluation of advanced engine technology, a reference mission and a reference airframe were defined representative of the year 2000 scenario and technology. The reference air-launched mission contained the following elements to enhance success:

- Long stand-off range,
- Low altitude terrain following,
- Dash phases, and
- Long range (2600 nautical miles (4815 km)).

Any engine configuration chosen for study had to perform the reference mission in the reference airframe. This airframe was fixed in cross section and allowed to grow in length to accommodate sufficient fuel for the mission. The reference airframe was defined through extrapolation of current trends in advanced materials, airframe aerodynamics, and other factors. The resulting airframe has a 22 percent lower weight and 29 percent lower drag coefficient than a current day missile of the same size. A current state-of-the-art turbofan engine was selected as a reference engine.

The second phase of the program was devoted to selecting an advanced engine cycle to power the advanced airframe. This work was initiated by conducting a parametric cycle study where primary variables of a turbofan engine were systematically varied. Component efficiency, which was defined as a function of component average flow parameter, was seen to decrease rapidly as component size decreased. This effect reduced the attractiveness of very high pressure ratio and high temperature cycles. This size effect on component efficiency suggested the possibility of a much lower pressure ratio recuperated cycle as a reasonable alternative to a conventional turbofan.

Two engines were selected for further study and preliminary definition layout. One was a 22:1 pressure ratio, 3.85 bypass ratio, 2200°F (1204°C) uncooled design. The other was a 2650°F (1454°C), 8:1 pressure ratio, 3.66:1 bypass ratio uncooled recuperated design with a 0.85 effectiveness. Both engine concepts were sized for the advanced reference missile defined in the Phase I study.

In the third phase of the program, a mission analysis was performed to compare missile systems powered by the reference engine, the advanced turbofan and the recuperated turbofan. Mission analysis was conducted using an existing terrain model. This mission, which was 2600 nautical miles (4815 km) long, was used to evaluate both the reference and advanced engine powered missile systems. Performance improvements were measured in terms of reductions in missile size and weight. When compared to the reference engine, the advanced turbofan engine powered missile exhibited a 38 percent reduction in fuel burned, resulting in a 25.3 percent reduction in length and 28.5 percent reduction in weight. The advanced recuperated turbofan powered missile exhibited a 47 percent reduction in fuel burned resulting in a 22.2 percent reduction in length and a 27.6 percent reduction in weight. The three engines were also evaluated in terms of mission flexibility using the following alternate launch/initial approach flight profiles.

- Air-launch/low level approach altitude,
- Surface-launch/optimum approach altitude, and
- Surface-launch/low level approach altitude.

The reference mission employed an air-launch/optimum approach altitude that required the minimum fuel to accomplish the 2600 nautical mile (4815 km) mission. Using the configurations that successfully accomplished the reference mission to fly the alternate mission profiles naturally resulted in shorter range. All engines exhibited a one percent range reduction for the surface-launch/optimum approach altitude profile. In the two low level approach profiles, the range reduction was on the order of six percent for all engines.

An engine life cycle cost (LCC) analysis was also carried out for each of the three engines in this phase of the program. The LCC of the advanced turbofan engine is 44 percent of the reference engine LCC, while the recuperated turbofan engine LCC is 53 percent of the reference engine LCC. Since the two advanced engines provide comparable performance for the reference mission and the alternate missions as well as yielding comparable missile system sizes and weights, LCC became the deciding criterion for selecting the advanced engine for the year 2000 cruise missile. Therefore, the advanced turbofan was selected over the recuperated turbofan on the basis of its 17 percent LCC superiority.

Finally, a technology plan was prepared outlining the programs required to provide the technology advances needed to realize the performance and LCC gains represented in the advanced turbofan engine. Ranked in the order of their contribution to the reduction in LCC, they are:

- Solid-Lubricated Bearing Program,
- Advanced Small Component Aerodynamics Program,
- Ceramic Composite Materials Program, and
- Slurry Fuel Technology Program.

TABLE I. REFERENCE ENGINE CHARACTERISTICS.

Mach 0.70, Sea Level	Characteristics With Flush Inlet	
Fn = 950 lb (4226 N)	OPR = 13.6	TIT = 1970°F (1077°C)
SFC = 0.987 lb/(lb-hr) (0.1007 kg/N • hr)	FPR = 2.1	Airflow = 29.7 lb/sec (13.5 kg/sec)
	BPR = 1.02	

TABLE II. REFERENCE ENGINE MATERIALS.

COMPONENT	MATERIAL
Fan	Titanium Alloy
Axial Compressor	Titanium Alloy
Centrifugal Compressor	Titanium Alloy
Combustor	Cobalt Alloy
HP Turbine	Nickel Superalloy
HP Turbine Nozzle	Cobalt Alloy
LP Turbine	Nickel Superalloy
Cold Structures	Aluminum Alloy
Hot Structures	Nickel Superalloy
Shaft	PH Stainless Steel

TABLE III. BASELINE ENGINE AERODYNAMIC COMPONENT CHARACTERISTICS.

COMPONENT	PRESSURE RATIO	EFFICIENCY (POLYTROPIC/ADIABATIC)	CORRECTED FLOW - LB/SEC (kg/s) IN	CORRECTED FLOW - LB/SEC (kg/s) OUT
2-stage fan (tip)	2.1	0.836/0.818	6.90 (3.13)	3.74 (1.70)
2-stage fan (hub)	2.1	0.875/0.861	6.75 (3.06)	3.56 (1.61)
2-stage axial compressor	1.68	0.845/0.834	3.56 (1.61)	2.31 (1.05)
centrifugal HP compressor	3.77	0.823/0.790	2.31 (1.05)	0.756 (0.343)
HP turbine	2.50	-/0.866	1.072 (0.486)	---
2-stage LP turbine	2.90	-/0.846	2.44 (1.11)	---

HP Shaft Speed = 64,275 RPM
LP Shaft Speed = 34,207 RPM

TABLE IV. ADVANCED TURBOFAN ENGINE MATERIALS.

COMPONENT	MATERIAL
Fan	Low Cost Net Shape Composite Resin
Axial Compressor	TI Aluminide
Centrifugal Compressor	TI Aluminide or Low Cost Net Shape Ceramic Composite
Combustor	Cobalt Alloy
HP Turbine	Low Cost High Temperature Net Shape Ceramic Composite
HP Turbine Nozzle	Superalloy
LP Turbine	Nickel Superalloy
Cold Structures	Short Fiber Reinforced Aluminum Composite
Hot Structures	Superalloy
Shaft	Metal Matrix

TABLE V. ADVANCED TURBOFAN ENGINE WEIGHT BREAKDOWN.

	WEIGHT LB (kg)
ROTATING STRUCTURE	30.5 (13.8)
FAN	3.3 (1.5)
IP STAGE ROTOR	2.2 (1.0)
HP TURBINE ROTOR	3.0 (1.4)
LP TURBINE ROTOR	12.1 (5.5)
SHAFT, LP	1.5 (0.7)
SHAFT, HP	1.1 (0.5)
CENTRIFUGAL COMPRESSOR ROTOR	5.2 (2.4)
BEARINGS	1.2 (0.5)
SPINNER	0.2 (0.1)
FUEL SLINGER	0.2 (0.1)
STATIC STRUCTURE	102.0 (46.3)
INLET	10.0 (4.5)
COMBUSTOR/NOZZLE	16.0 (7.3)
INTERSTAGE HOUSING	27.0 (12.2)
2ND NOZZLE	17.0 (7.7)
REAR HOUSING	10.0 (4.5)
BYPASS DUCT	10.0 (4.5)
DIFFUSER	12.0 (5.4)
ACCESSORIES	32.5 (14.7)
FUEL CONTROL	17.0 (7.7)
IGNITION SYSTEM	9.0 (4.1)
WIRE HARNESS	3.5 (1.6)
INSTRUMENTATION	3.0 (1.4)
TOTAL DRY ENGINE	165 (74.8)
FLUIDS, FUEL MANAGEMENT SYSTEM, AIRFRAME INLET DUCT	49 (22.2)
TOTAL PROPULSION SYSTEM	214 (97.1)

TABLE VI. ADVANCED TURBOFAN ENGINE AERODYNAMIC COMPONENT CHARACTERISTICS.

COMPONENT	PRESSURE RATIO	EFFICIENCY (POLYTROPIC/ADIABATIC)	CORRECTED FLOW - LB/SEC (kg/s)	
			IN	OUT
Single Stage Fan (tip)	1.7	0.901/0.894	26.24 (11.90)	16.79 (7.62)
Single Stage Fan (hub)	1.7	0.901/0.894	6.82 (3.09)	4.36 (1.98)
Single Stage Axial Compressor	1.8	0.889/0.880	4.36 (1.98)	2.66 (1.21)
Centrifugal Compressor	7.19	0.886/0.853	2.66 (1.21)	0.498 (0.226)
Single Stage HP Turbine	4.3	-/0.890	0.704 (0.319)	---
1 1/2 Stage LP Turbine	3.0	-/0.926	2.61 (1.18)	---

HP Shaft Speed = 53,170 RPM
LP Shaft Speed = 22,330 RPM

TABLE VII. RECUPERATED TURBOFAN ENGINE MATERIALS.

COMPONENT	MATERIAL
Fan	Low Cost Net Shape Composite Resin
Centrifugal Compressor	Titanium or Low Cost Net Shape Ceramic Composite
Combustor	Composite
HP Turbine	Low Cost High Temperature Net Shape Ceramic Composite
HP Turbine Nozzle	Structured Ceramic Composite
Cold Structures	Short Fiber Reinforced Aluminum Composite
Hot Structures	Structured High Temperature Composite
Shaft	Metal Matrix

TABLE VIII. RECUPERATED TURBOFAN ENGINE WEIGHT BREAKDOWN.

	WEIGHT LB (kg)
ROTATING STRUCTURE	27.5 (12.5)
FAN	4.0 (1.8)
TURBINE ROTOR	3.8 (1.7)
CENTRIFUGAL COMPRESSOR ROTOR	6.5 (2.9)
GEAR REDUCTION	8.2 (3.7)
SHAFTS	2.0 (0.9)
AIR BEARING	1.5 (0.7)
BALL BEARING	0.5 (0.2)
FUEL SLINGER	0.5 (0.2)
SPINNER	0.5 (0.2)
STATIC STRUCTURE	89.0 (40.4)
INLET	12.5 (5.7)
INTERSTAGE HOUSING	33.0 (15.0)
DIFFUSER	15.0 (6.8)
COMBUSTOR/NOZZLE	8.5 (3.9)
REAR HOUSING	5.0 (2.3)
BYPASS DUCT	15.0 (6.8)
ACCESSORIES	39.5 (17.9)
FUEL CONTROL	17.0 (7.7)
IGNITION SYSTEM	9.0 (4.1)
WIRE HARNESS	3.5 (1.6)
INSTRUMENTATION	3.0 (1.4)
OIL SYSTEM (FOR GEARBOX)	7.0 (3.2)
RECUPERATOR CORE, VARIABLE GEOMETRY EXHAUST HARDWARE AND DUCTS	240.0 (108.9)
TOTAL DRY ENGINE	396 (179.6)
FLUIDS, FUEL MANAGEMENT SYSTEM, AIRFRAME INLET DUCT	49 (22.2)
TOTAL PROPULSION SYSTEM	445 (201.8)

TABLE IX. RECUPERATED TURBOFAN ENGINE AERODYNAMIC COMPONENT CHARACTERISTICS.

SEA LEVEL, MACH 0.7, STANDARD DAY OPERATING POINT				
COMPONENT	PRESSURE RATIO	EFFICIENCY (POLYTROPIC/ADIABATIC)	CORRECTED FLOW - LB/SEC (kg/s) IN	OUT
Single Stage Geared Fan (tip)	1.7	0.901/0.894	25.78 (11.69)	16.49 (7.48)
Single Stage Geared Fan (hub)	1.7	0.901/0.894	7.04 (3.19)	4.58 (2.08)
Centrifugal Compressor	4.7	0.894/0.870	4.51 (2.05)	1.245 (0.565)
Single Stage Turbine	3.6	-/0.920	2.24 (1.016)	—
GAS GENERATOR SHAFT SPEED = 55,000 RPM FAN SHAFT SPEED = 30,136 RPM				

TABLE X. MISSION PERFORMANCE.

	ENGINE	AIR LAUNCH OPT ALT APPROACH	AIR LAUNCH LOW LEVEL APPROACH	SURFACE LAUNCH OPT ALT APPROACH	SURFACE LAUNCH LOW LEVEL APPROACH
APPROACH CRUISE ALTITUDE: FT (m)	REFERENCE	18500 (5639)	100 (30)	19500 (5944)	100 (30)
	ADV TURBOFAN	20000 (6096)	100 (30)	20000 (6096)	100 (30)
	ADV RECUP	20000 (6096)	100 (30)	18500 (5639)	100 (30)
RANGE: N. MILES (km)	REFERENCE	2600 (4815)	2453 (4543)	2578 (4774)	2428 (4497)
	ADV TURBOFAN	2600 (4815)	2440 (4519)	2577 (4773)	2413 (4469)
	ADV RECUP	2600 (4815)	2459 (4554)	2576 (4771)	2434 (4508)

TABLE XI. REFERENCE ENGINE MISSILE SIZE AND WEIGHT REQUIREMENTS.

CASE	ENGINE SIZE: SL MACH 0.7 THRUST Lb (N)	GROSS WT. Lb (kg)	FUEL WT. Lb (kg)	LENGTH Ft (m)	WING AREA Ft^2 (m^2)	RELATIVE CLIMB CAPABILITY
1	966 (4297)	5055 (2293)	3245 (1472)	35.6 (10.9)	23.6 (2.2)	1.0
2	966 (4297)	4063 (1843)	2446 (1110)	29.3 (8.9)	19.0 (1.8)	1.35
3	782 (3478)	3838 (1741)	2265 (1027)	27.9 (8.5)	17.9 (1.7)	1.0

TABLE XII. SECT MISSION FUEL CONSUMPTION CHARACTERISTICS.

MISSION ELEMENT	ENGINE	PERCENT FUEL LOAD CONSUMED			
		AIR LAUNCH		SURFACE LAUNCH	
		OPT. ALT. APPROACH	LOW LEVEL APPROACH	OPT. ALT. APPROACH	LOW LEVEL APPROACH
APPROACH	REFERENCE	18.5	24.9	19.5	26.1
	ADV. TURBOFAN	17.8	24.9	18.8	26.0
	ADV. RECUP.	18.0	24.2	19.0	25.4
TERRAIN FOL. #1	REFERENCE	13.2	13.0	13.2	12.9
	ADV. TURBOFAN	12.9	12.7	12.9	12.7
	ADV. RECUP.	13.1	12.9	13.0	12.9
TERRAIN FOL. #2	REFERENCE	44.1	43.7	44.0	43.6
	ADV. TURBOFAN	44.5	44.1	44.5	44.0
	ADV. RECUP.	43.9	43.5	43.8	43.4
TERRAIN FOL. #3	REFERENCE	19.3	13.5	18.5	12.5
	ADV. TURBOFAN	19.6	13.2	18.8	12.2
	ADV. RECUP.	19.5	13.9	18.7	12.8
3 DASH PHASES	REFERENCE	4.9	4.9	4.8	4.9
	ADV. TURBOFAN	5.1	5.1	5.1	5.1
	ADV. RECUP.	5.5	5.5	5.5	5.5
RANGE: N. M. (km)	REFERENCE	2600 (4815)	2453 (4543)	2578 (4774)	2428 (4497)
	ADV. TURBOFAN	2600 (4815)	2440 (4519)	2577 (4773)	2413 (4469)
	ADV. RECUP.	2600 (4815)	2459 (4554)	2576 (4771)	2434 (4508)

TABLE XIII. ENGINE TURBINE INLET TEMPERATURE.

TIT: °F (°C) AVG/PEAK*			
ENGINE	LOW ALTITUDE MACH 0.7 CRUISE	MAXIMUM DURING FIRST ACCEL TO MACH 0.9	DURING FIRST MACH 0.9 CRUISE
REFERENCE	1595/1679 (868/915)	2013/2185 (1101/1196)	1744/1893 (951/1034)
ADVANCED TURBOFAN	1764/1900 (962/1038)	2222/2396 (1217/1313)	2010/2165 (1099/1185)
ADVANCED RECUPERATED	2138/2243 (1170/1229)	2650/2787 (1454/1530)	2422/2545 (1328/1396)

*PATTERN FACTOR = 0.15

TIT_{PEAK} = (PATTERN FACTOR)(TIT_{AVG} - $T_{(COMBUSTOR\ INLET)}$) + TIT_{AVG}

TABLE XIV. SECT MISSILE SIZE AND WEIGHT.
(DIAMETER = 20.38 INCHES (51.77 cm))

	REFERENCE	ENGINE ADVANCED TURBOFAN	ADVANCED RECUPERATED
GROSS WEIGHT (lb (kg))	4063 (1843)	2906 (1318)	2940 (1334)
FUEL BURNED (lb (kg))	2446 (1110)	1510 (685)	1291 (586)
PROPULSION SYSTEM* WEIGHT (lb (kg))	214 (97)	214 (97)	445 (202)
ENGINE WEIGHT (lb (kg))	165 (75)	165 (75)	396 (180)
RECUPERATOR RELATED WEIGHT** (lb (kg))	-	-	240 (109)
ENGINE LENGTH (ft (m))	3.1 (0.9)	3.1 (0.9)	5.7 (1.7)
EFF. FUEL TANK LENGTH (ft (m))	19.3 (5.9)	11.9 (3.6)	10.2 (3.1)
MISSILE LENGTH (ft (m))	29.3 (8.9)	21.9 (6.7)	22.8 (6.9)
WING AREA (sq ft (m^2))	19.0 (1.8)	13.6 (1.3)	13.7 (1.3)

*INCLUDES INLET DUCT, FUEL MANAGEMENT SYSTEM, ETC.

**INCLUDES VARIABLE GEOMETRY EXHAUST HARDWARE, RECUPERATOR CORE, AND DUCTS

TABLE XV. SUMMARY OF ADVANCED TECHNOLOGY ENGINE BENEFITS.

		REFERENCE MISSILE WITH ADVANCED ENGINE COMPARED TO REFERENCE ENGINE
GROSS WEIGHT: lb (kg)		
REFERENCE ENGINE	4063 (1843)	-
ADVANCED TURBOFAN	2906 (1318)	-28.5%
ADVANCED RECUPERATED	2940 (1334)	-27.6%
FUEL BURNED: lb (kg)		
REFERENCE ENGINE	2446 (1109)	-
ADVANCED TURBOFAN	1510 (685)	-38.3%
ADVANCED RECUPERATED	1291 (586)	-47.2%
MISSILE LENGTH: ft (m)		
REFERENCE ENGINE	29.3 (8.9)	-
ADVANCED TURBOFAN	21.9 (6.7)	-25.3%
ADVANCED RECUPERATED	22.8 (6.9)	-22.2%
WING AREA: sq ft (m^2)		
REFERENCE ENGINE	19.0 (1.8)	-
ADVANCED TURBOFAN	13.6 (1.3)	-28.5%
ADVANCED RECUPERATED	13.7 (1.3)	-27.6%

TABLE XVI. EFFECT OF USING SF-2 CARBON SLURRY.

		REFERENCE MISSILE WITH SF-2 CARBON SLURRY COMPARED TO JP-10
GROSS WEIGHT: lb (kg)		
REFERENCE ENGINE	4495 (2039)	+10.6%
ADVANCED TURBOFAN	3073 (1394)	+ 5.7%
ADVANCED RECUPERATED	3071 (1393)	+ 4.5%
FUEL BURNED: lb (kg)		
REFERENCE ENGINE	2916 (1323)	+19.2%
ADVANCED TURBOFAN	1717 (779)	+13.7%
ADVANCED RECUPERATED	1458 (661)	+12.9%
MISSILE LENGTH: ft (m)		
REFERENCE ENGINE	27.2 (8.3)	- 7.2%
ADVANCED TURBOFAN	20.1 (6.1)	- 8.2%
ADVANCED RECUPERATED	21.2 (6.5)	- 7.0%
WING AREA: sq ft (m^2)		
REFERENCE ENGINE	21.0 (2.0)	+10.6%
ADVANCED TURBOFAN	14.4 (1.3)	+ 5.7%
ADVANCED RECUPERATED	14.4 (1.3)	+ 4.5%

TABLE XVII. ADDITIONAL BENEFITS OF SF-2 SLURRY FUEL.

- TO FLY THE 2600-NAUTICAL MILE (4815 km) REFERENCE MISSION; MISSILE LENGTH IS REDUCED:

ENGINE	MISSILE LENGTH: ft (m)		MISSILE LENGTH CHANGE
	JP-10	SF-2	
REFERENCE	29.3 (8.93)	27.2 (8.29)	-7.2%
ADV TURBOFAN	21.9 (6.68)	20.1 (6.13)	-8.2%
ADV RECUPERATED	22.8 (6.95)	21.2 (6.46)	-7.0%

- FOR THE SAME MISSILE LENGTH USED WITH JP-10 FOR THE 2600-NAUTICAL MILE (4815 km) MISSION:

ENGINE	MISSILE LENGTH ft (m)	RANGE N. MILES (km)	RANGE CHANGE %
REFERENCE	29.3 (8.93)	2775 (5139)	+6.7
ADV TURBOFAN	21.9 (6.68)	2855 (5287)	+9.8
ADV RECUPERATED	22.8 (6.95)	2867 (5310)	+10.3

TABLE XVIII. UNIT LIFE CYCLE COST ESTIMATES (20 YEARS).

	PERCENT OF REFERENCE ENGINE LCC			
	REFERENCE ENGINE 2 MAJOR & & 2 MINOR RECERTIFICATION	REFERENCE ENGINE WITH ONLY 1 MAJOR RECERT*	ADVANCED TURBOFAN	ADVANCED RECUPERATED TURBOFAN
DEVELOPMENT	9.3	9.3	4.9	7.0
PRODUCTION	37.2	37.2	12.4	16.8 (2.1 RECUP) (2.3 GEARED FAN)
OPERATING & SUPPORT (O&S)				
MAJOR RECERT LABOR	5.8	2.9	2.5	2.6
MAJOR RECERT MATL	7.2	3.6	1.2	1.6
MINOR RECERT LABOR	1.5	-	-	-
MINOR RECERT MATL	4.3	-	-	-
REFURB.	21.6	21.6	11.6	13.6
FUEL/OIL	2.0	0.5	0.3	0.3
GRD SUPPORT EQUIP.	2.3	2.3	2.3	2.3
MISC.	8.8	8.8	8.8	8.8
TOTAL O&S	53.5	39.7	26.7	29.2
TOTAL LCC	100.0	86.2	44.0	53.0

*SHOWN TO ILLUSTRATE EFFECT OF REDUCTION IN RECERTIFICATIONS ALONE. (NOT FEASIBLE WITH TODAY'S STATE OF THE ART)

TABLE XIX. UNIT PRODUCTION COST.

REFERENCE ENGINE	ADVANCED TURBOFAN	ADVANCED RECUPERATED TURBOFAN
● 2-STAGE FAN	● SINGLE STAGE FAN	● SINGLE STAGE FAN
● 2-STAGE IP	● SINGLE IP STAGE	● SINGLE STAGE CENT. CPRSR
● 1-STAGE CENT. CPRSR	● SINGLE CENT. CPRSR	● SINGLE STAGE TURBINE
● 1-STAGE HP TURBINE	● SINGLE HP TURBINE	● 4 MAIN SHAFT BRGS (1 AIR BRG)
● 2 STAGE LP TURBINE	● SINGLE LP TURBINE	● DRY FILM LUBRICANT
● 6 MAIN SHAFT BEARINGS	● 4 MAIN SHAFT BEARINGS	● SIMPLIFIED FUEL SYSTEM
● FULL LUBRICATION SUBSYSTEM	● DRY FILM LUBRICANT	● FAN REDUCTION GEAR
● GEARBOX ASSY	● SIMPLIFIED FUEL SYSTEM	● RECUPERATOR
● FULL FUEL SUBSYSTEM	● NO GEARBOX	● PYRO-IGNITION SYSTEM
● ELECTRICAL IGNITION SYSTEM	● PYRO-IGNITION SYSTEM	
PUC* = 100.0	PUC = 33.3	PUC = 45.1
*PERCENT OF REFERENCE ENGINE PRODUCTION UNIT COST		

TABLE XX. PRODUCT UNIT COST ESTIMATES.

COMPONENT	REFERENCE ENGINE (PUC*)	ADVANCED TURBOFAN (PUC*)	ADV RECUPERATED TURBOFAN (PUC*)
FAN STAGE ROTOR	7.4	3.7	3.7
IP STAGE ROTOR	7.4	3.7	-
CENT. COMPRESSOR ROTOR	7.0	7.0	7.0
HP TURBINE ROTOR	4.0	4.0	4.0
LP TURBINE ROTOR	3.2	1.4	-
STATIONARY STRUCTURES	30.0	9.8	15.0
LUBRICATION SYBSYS (EXTERNAL)	5.5	-	2.0
GEARBOX ASSY	24.8	-	4.0
FUEL SUBSYSTEMS	8.9	3.0	3.0
IGNITION SUBSYSTEMS	1.8	0.7	0.7
RECUPERATOR	-	-	5.7
	100.0	33.3	45.1

*PERCENT OF REFERENCE ENGINE PRODUCTION UNIT COST. THESE ESTIMATES ARE BASED ON MID-POINT OF 5000 UNIT PRODUCTION RUN.

TABLE XXI. TECHNOLOGY PLAN SUMMARY.

PROGRAM	OBJECTIVES	PAYOFFS*
Solid Lubricated Bearing Program	• Demonstrate Advanced Bearing Hardware and Solid Lubricant	• Reduce Cost Small Turbofan Engine • Accounts for 45.43% of the Total LCC Reduction
Advanced Small Component Aerodynamics Program	• Identify, Quantify and Control the Influence of Secondary Flows	• Improved Component Efficiency (3-5%) • Accounts for 24.2% of the Total LCC Reduction
Ceramic Composite Materials Program	• Demonstrate High Temperature, High Speed Turbine Operation Without Cooling • Establish Domestic Technology for Composite Materials	• Improved Small Turbofan Performance • Accounts for 22.58% of the Total LCC Reduction
Slurry Fuel Technology Program	• Demonstrate High Efficiency Small Combustion System Using Slurry Fuels	• Improved Missile System Range (10%)

*Additional LCC Reductions Attributable to Simplification of Fuel and Ignition Systems.

TABLE XXII. DIMENSIONAL VARIATIONS FOR A TYPICAL GEOMETRIC VARIABLE.

(EXAMPLE: AXIAL COMPRESSOR ROTOR BLADE FILLET)

TYPICAL CRUISE MISSILE VALUE MFG VARIATION	UPSCALE TEST VALUE MFG VARIATION	UPSCALE TEST VARIATION
0.050 (±0.005) in. 0.127 (±0.013) cm (±10%)	0.125 (±0.005) in. 0.318 (±0.013) cm (±4%)	0.125 (±0.050) in. 0.318 (±0.130) cm (±40%)

(LINEAR SCALE FACTOR = 2.5:1)

TABLE XXIII. CRITICAL UPSCALED GEOMETRIC VARIABLES.

(EXAMPLE: CENTRIFUGAL COMPRESSOR)

ITEM	TYPICAL CRUISE MISSILE VALUE/% VARIATION	UPSCALE TEST VALUE/% TEST VARIATION	UPSCALE TEST VALUE/% MFG VARIATION
ROTOR TIP WIDTH: IN. (cm)	0.290 ±0.001/±0.3% (0.737 ±0.003/±0.3%)	0.725/±? (1.842/±?)	0.725 ±0.002/±0.3% (1.842 ±0.005/±0.3%)
ROTOR BLADE FILLET: IN. (cm)	0.098 ±0.005/±5.1% (0.249 ±0.013/±5.1%)	0.245/±100% (0.622/±100%)	0.245 ±0.005/± 2% (0.622 ±0.013/±2%)
ROTOR BLADE THICKNESS: IN. (cm)	0.048 ±0.003/±6.3% (0.122 ±0.008/±6.3%)	0.120/±50% (0.305/±50%)	0.120 ±0.003/±2.5% (0.305 ±0.008/±2.5%)
ROTOR THROAT AREA: $IN.^2$ (cm^2)	0.30 ±0.005/±1.7% (1.935 ±0.032/±1.7%)	1.875/±6% (12.097/±6%)	1.875 ±0.008/±0.4% (12.097 ±0.052/±0.4%)
RUNNING CLEARANCE: IN. (cm)	0.008 ±0.002/±25% (0.020 ±0.005/±25%)	0.020/± 50% (0.051/±50%)	0.020 ±0.004/± 20% (0.051 ±0.010/±20%)
SURFACE FINISH	32 RMS/±?	80 RMS/±80%	80 RMS/±?
RADIAL VANE LER: IN. (cm)	0.007 ±0.002/± 29% (0.018 ±0.005/±29%)	0.0175/±50% (0.0445/±50%)	0.0175 ±0.002/±11.4% (0.0445 ±0.005/±11.4%)
RADIAL VANE THROAT: $IN.^2$ (cm^2)	0.1057/±1.6% (0.6819/±1.6%)	0.661/±6% (4.265/±6%)	0.661 ±0.008/±1.1% (4.265 ±0.052/±1.1%)
DESWIRL VANE TER: IN. (cm)	0.0075 ±0.0025/±33% (0.0191 ±0.0064/±33%)	0.0188/±50% (0.0478/±50%)	0.0188 ±0.025/±13% (0.0478 ±0.064/±13%)
DESWIRL VANE LER: IN. (cm)	0.010 ±0.001/±10% (0.025 ±0.003/±10%)	0.025/±50% (0.064/±50%)	0.025 ±0.001/±4% (0.064 ±0.003/±4%)
DESWIRL VANE CHORD: IN. (cm)	0.800 ±0.005/±6% (2.032 ±0.013/±6%	2.0/±10% (5.08/±10%)	2.0 ±0.005/±0.25% (5.08 ±0.013/±0.25%)
DESWIRL VANE T/C	0.10/±5.6%	0.10/±50%	0.10/±2.2%
DESWIRL VANE: DEG (mrad)	20 ±0.5/±2.5% (349 ±8.7/±2.5%)	20/±20% (349 ±20%)	20 ±0.25/±1.25% (349 ±4.36/±1.25%)

LINEAR SCALE FACTOR = 2.5:1

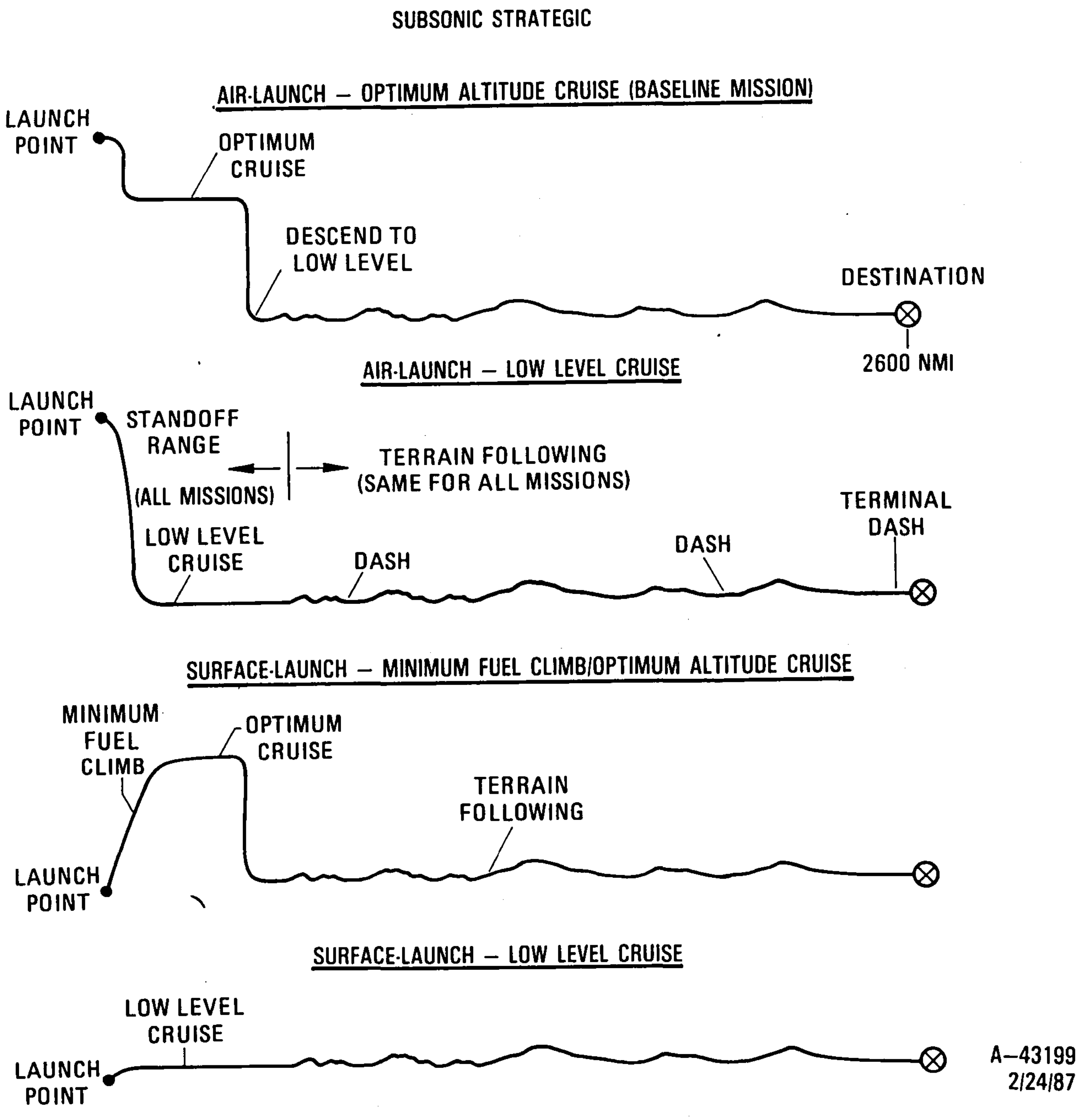

Figure 1. Reference Mission Variants.

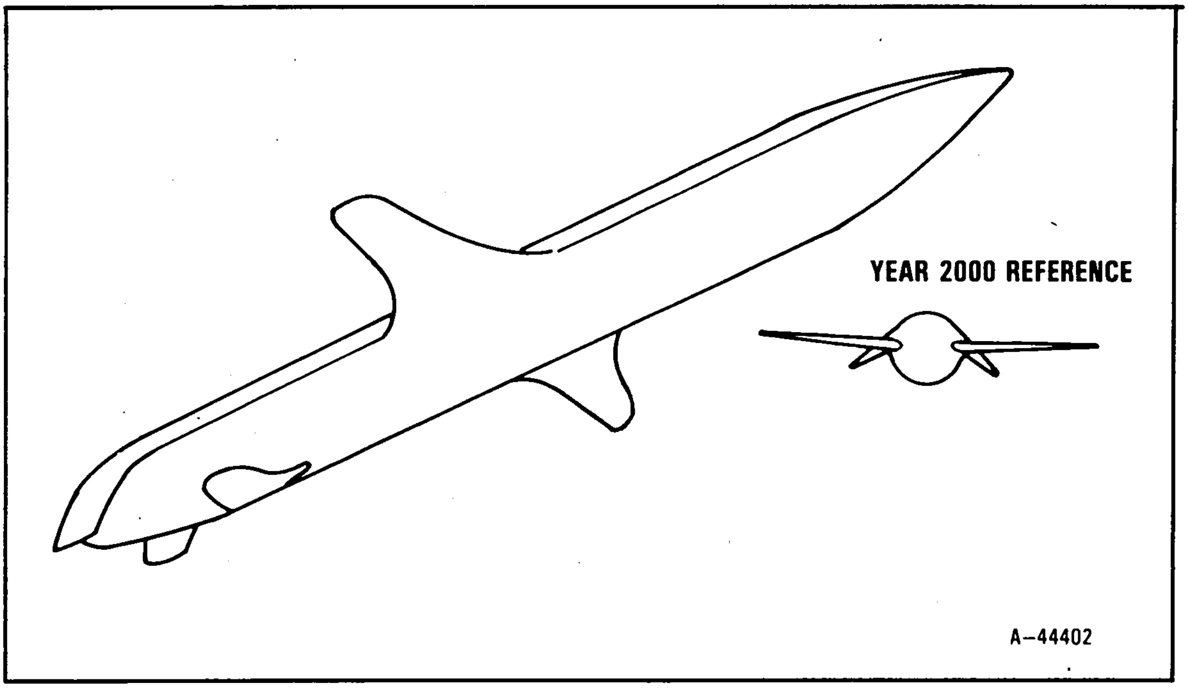

Figure 2. Airframe Configurations.

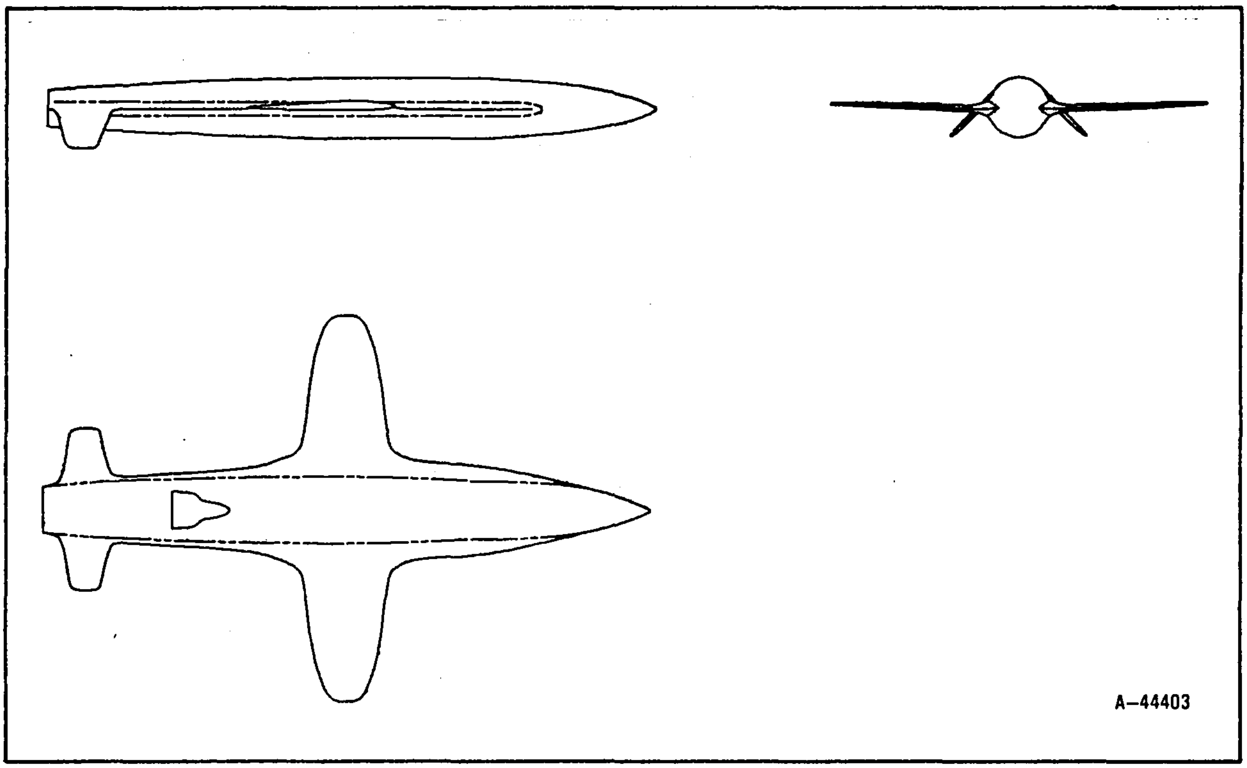

Figure 3. Reference Year 2000 Subsonic Strategic Cruise Missile Configuration.

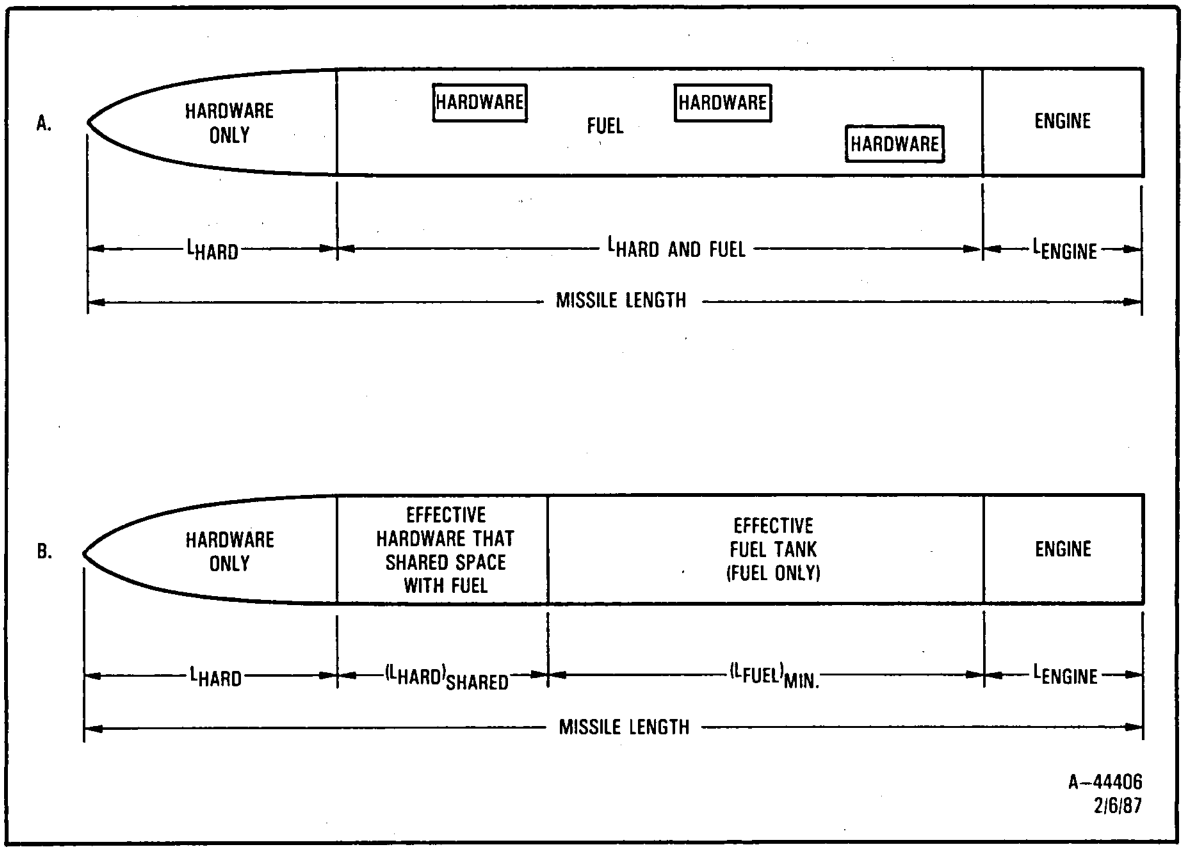

Figure 4. Effective Fuel Tank Length.

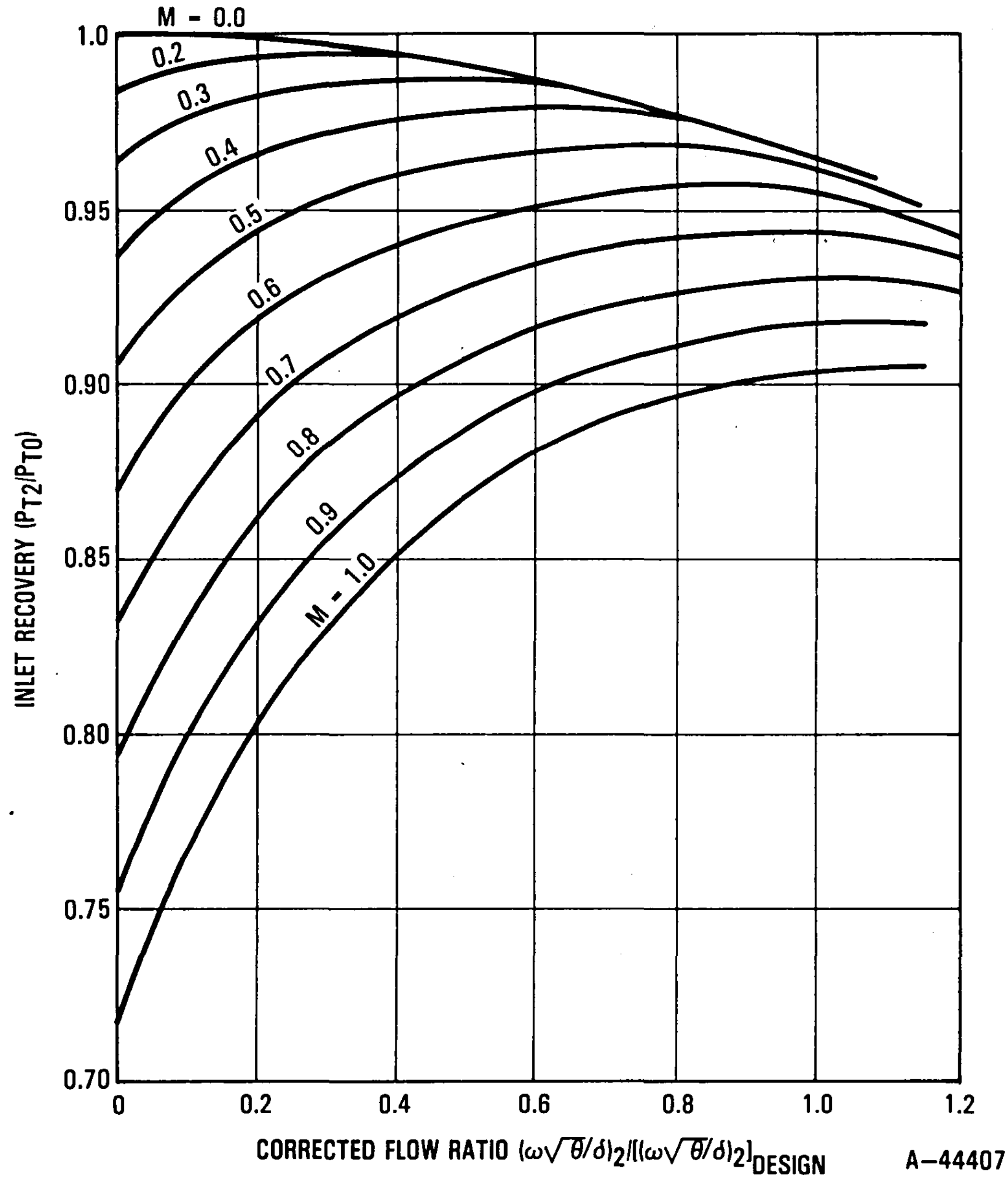

Figure 5. Flush Inlet Recovery Characteristics.

MISSION REQUIREMENTS	SURVIVABILITY		AVAILABILITY	
SUCCESS CONTRIBUTORS	PERFORMANCE	SIZE	COST	RELIABILITY
SUCCESS QUANTIFIERS	SPECIFIC FUEL CONSUMPTION SPECIFIC THRUST TRANSIENT RESPONSE	WEIGHT DIAMETER VOLUME	HARDWARE DESIGN/DEVELOPMENT LIFE CYCLE COST	TEST RESULTS FAILURES FIELD EXPERIENCE
TRADE FACTORS	PRESSURE RATIO TURBINE TEMPERATURE COMPONENT EFFICIENCIES HEAT RECOVERY	CONFIGURATION ROTATIONAL SPEED COMPONENT TYPE	MANUFACTURING PROCEDURE MATERIALS TECHNICAL RISK	COMPLEXITY LOADINGS DESIGN APPROACH

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Figure 6. Cruise Missile Engine Evaluation Procedure.

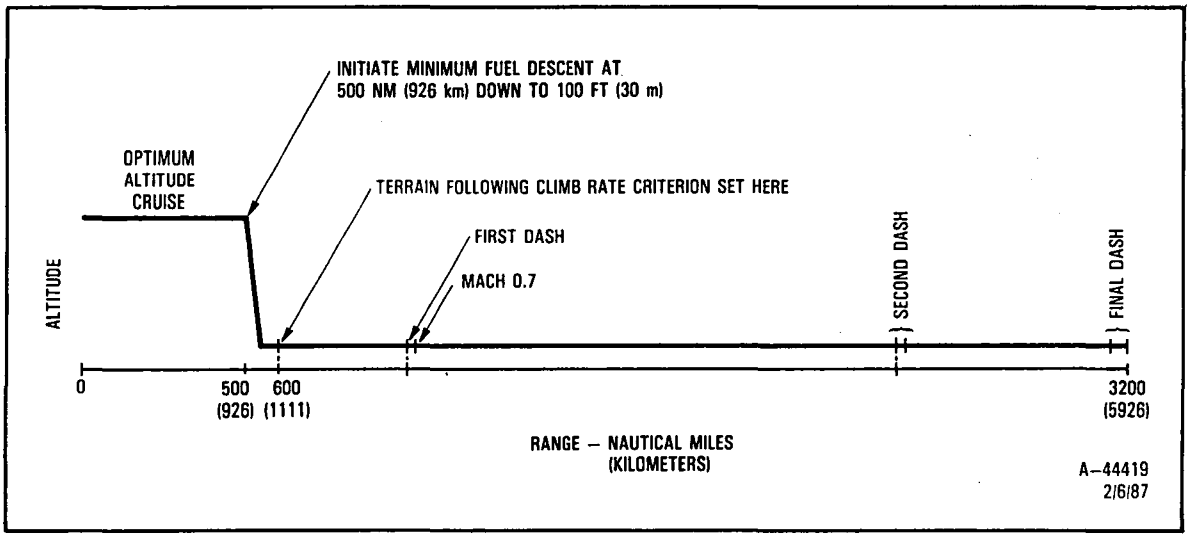


Figure 7. Propulsive Range Mission Profile.

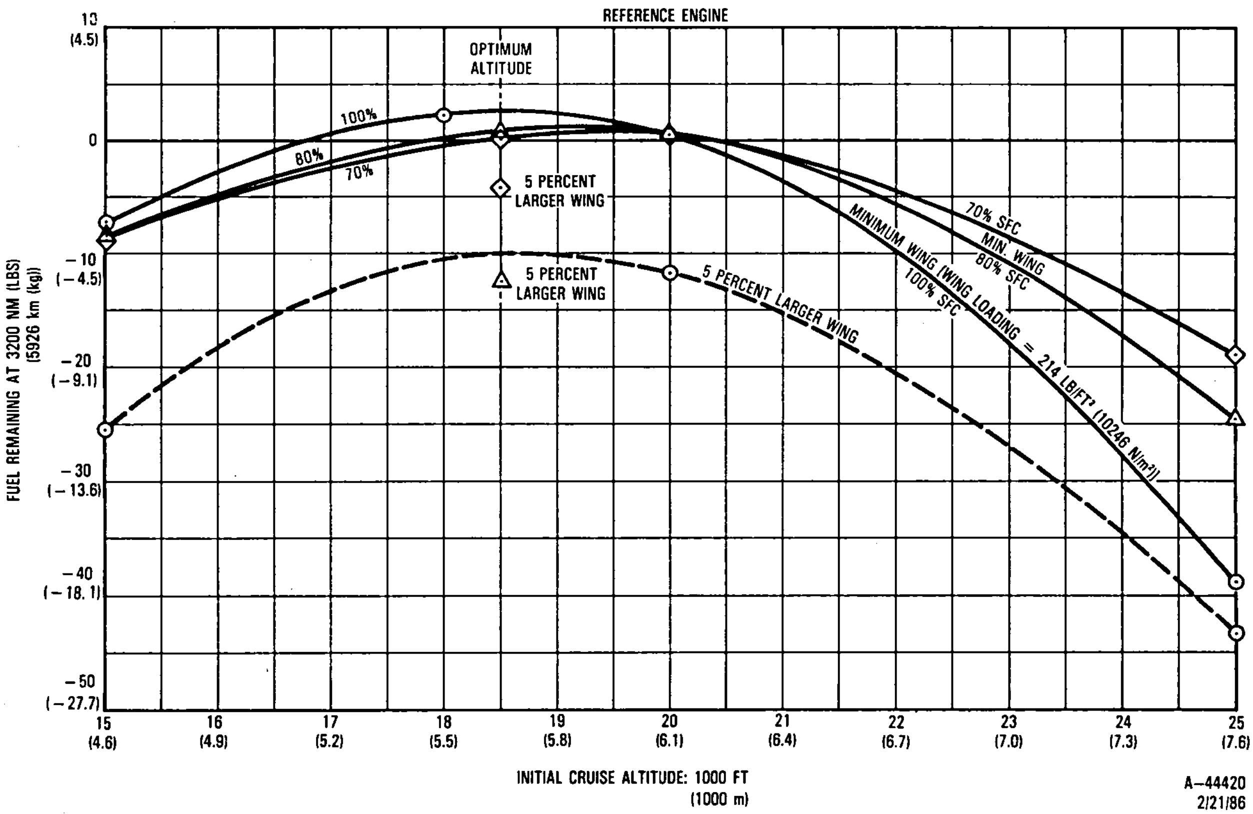
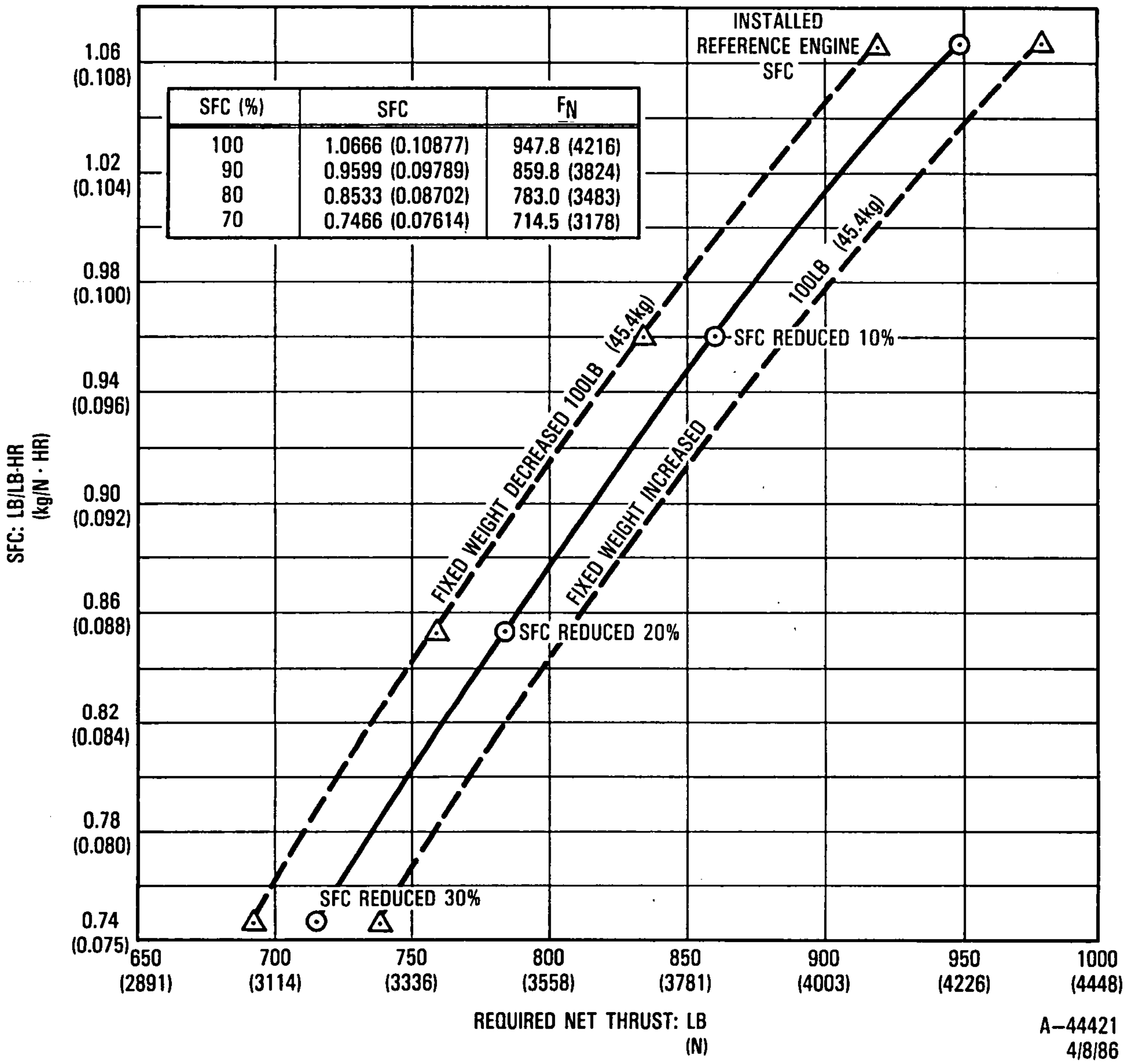


Figure 8. Optimizing Initial Cruise Altitude and Wing Size.



SFC (%)	SFC	F_N
100	1.0666 (0.10877)	947.8 (4216)
90	0.9599 (0.09789)	859.8 (3824)
80	0.8533 (0.08702)	783.0 (3483)
70	0.7466 (0.07614)	714.5 (3178)

Figure 9. Effect of SFC on Required Thrust.

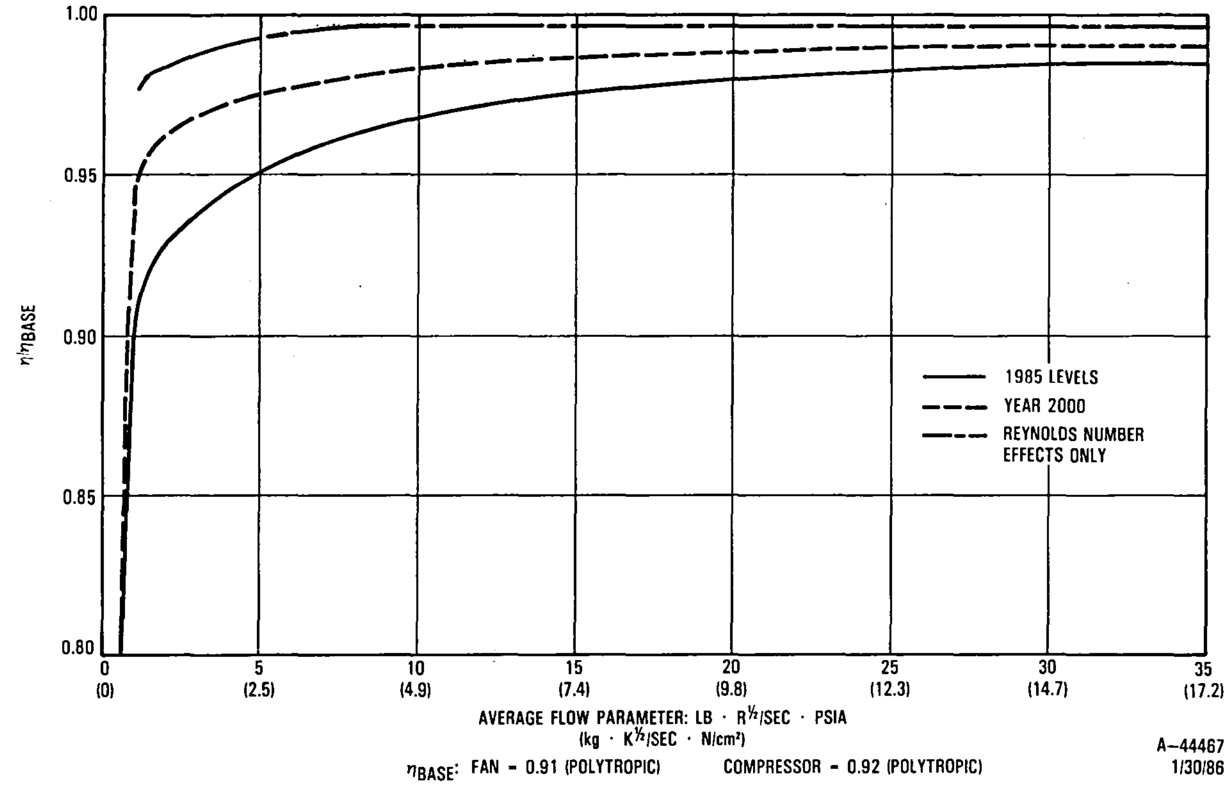


Figure 10. Fan and Compressor Efficiency Trends.

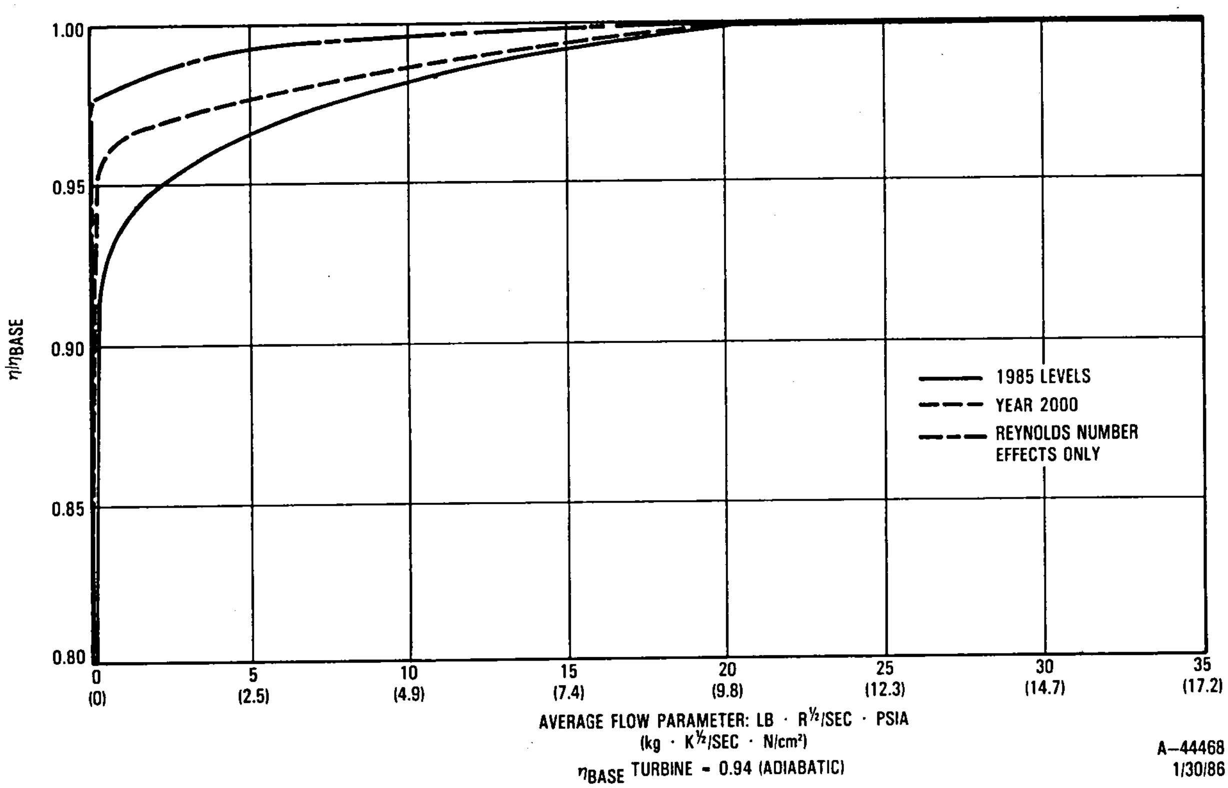


Figure 11. Turbine Efficiency Trends.

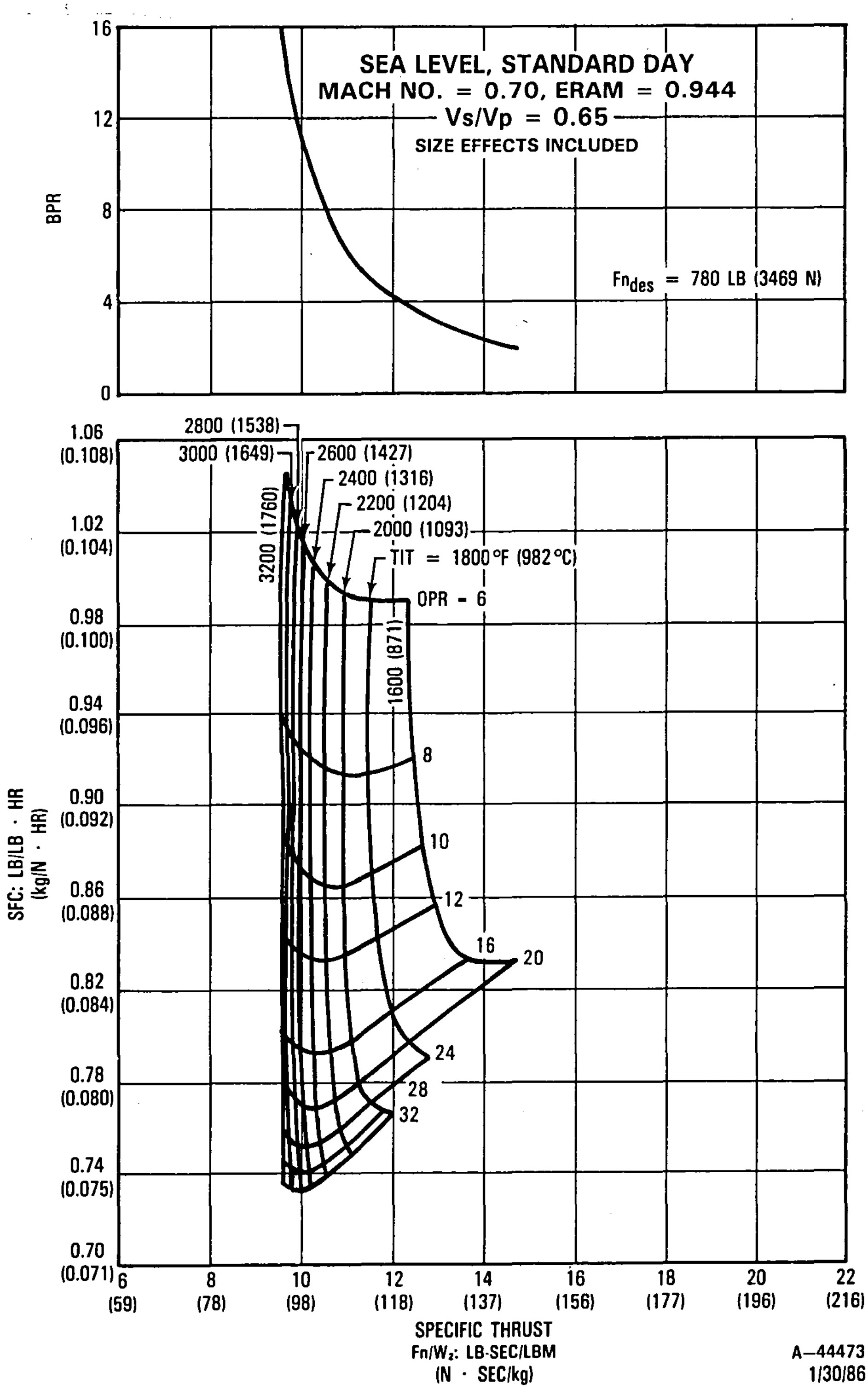


Figure 12. Advanced Turbofan Engine (Fan PR = 1.4).

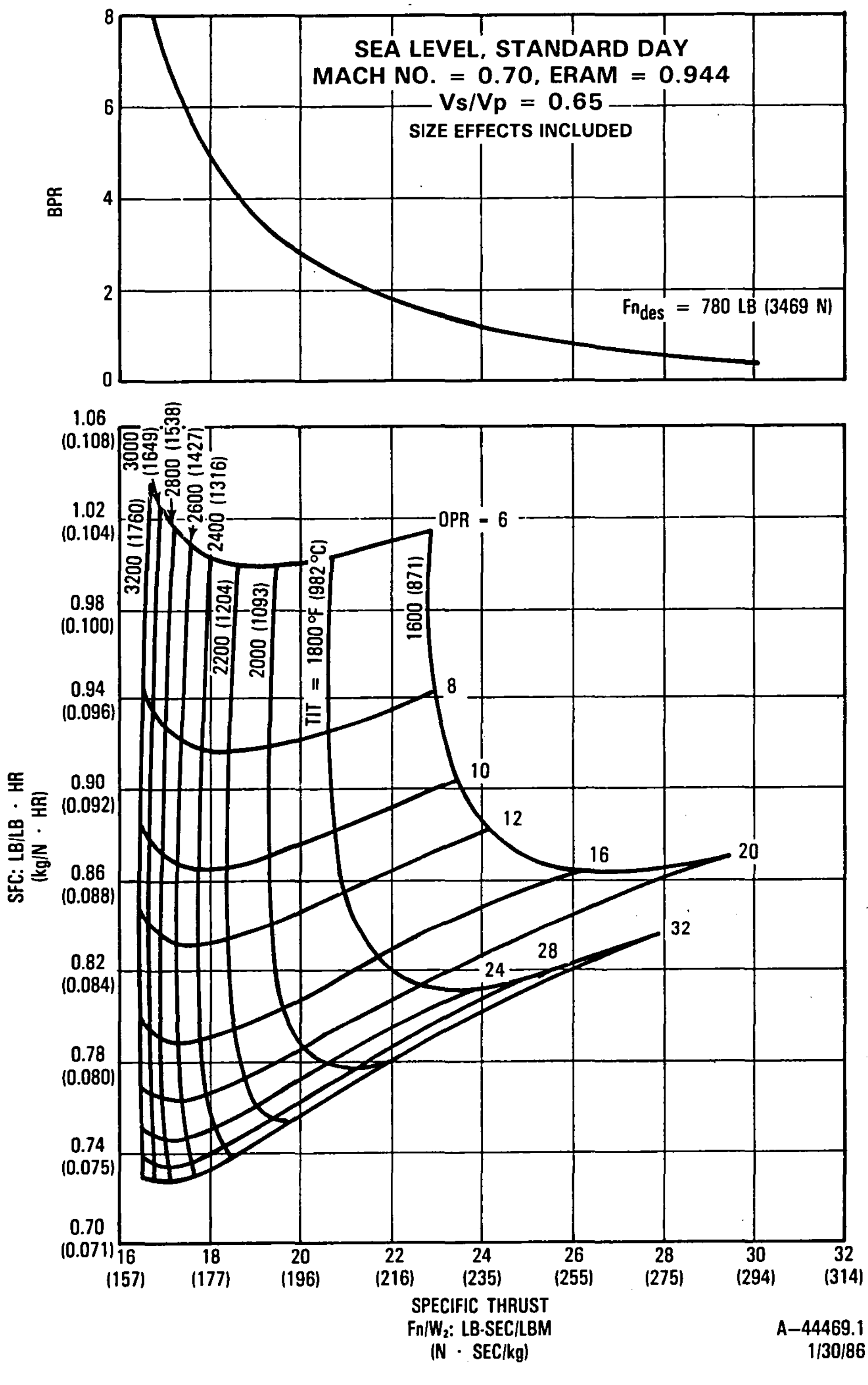


Figure 13. Advanced Turbofan Engine (Fan PR = 1.7).

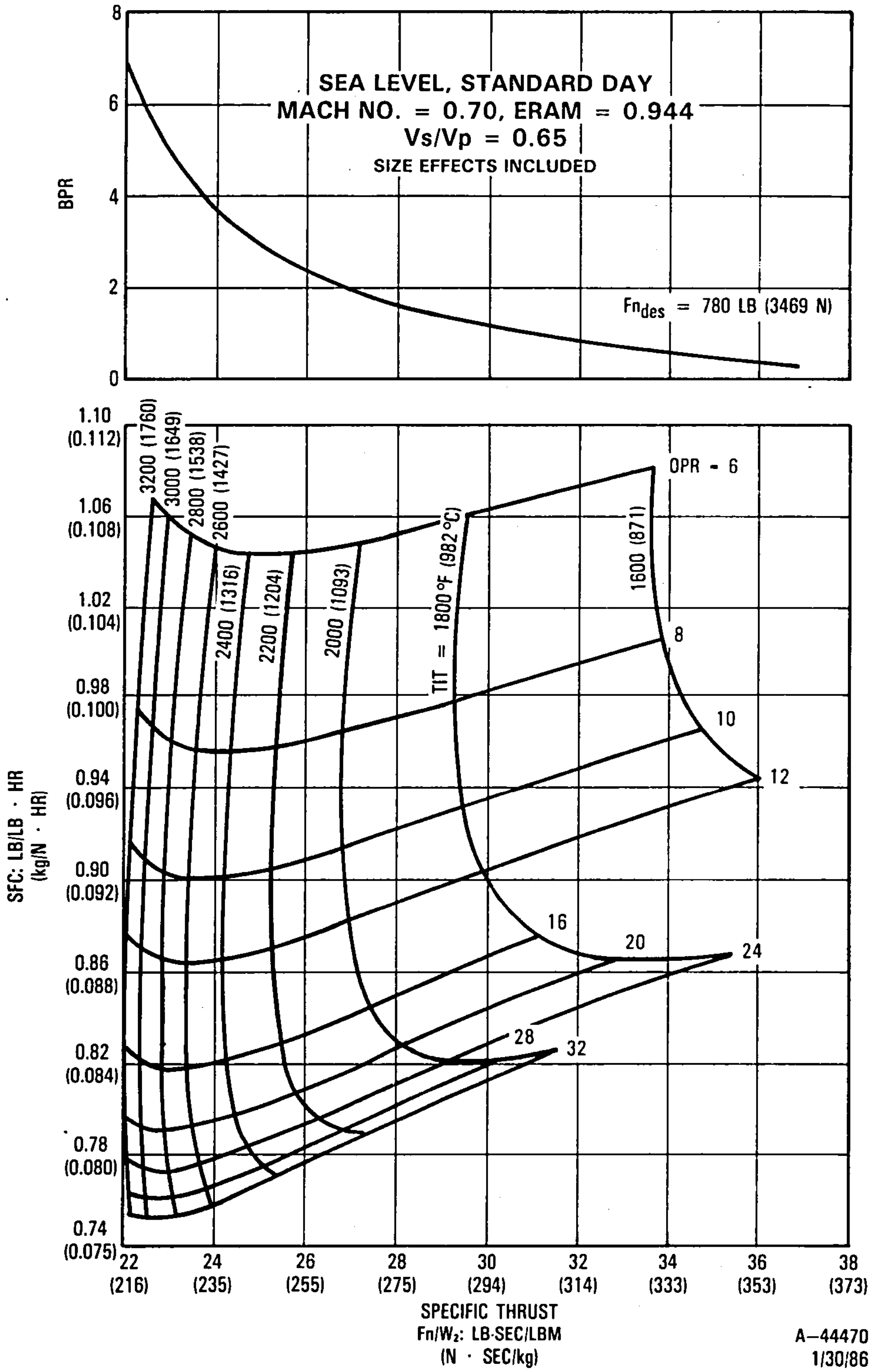


Figure 14. Advanced Turbofan Engine (Fan PR = 2.0).

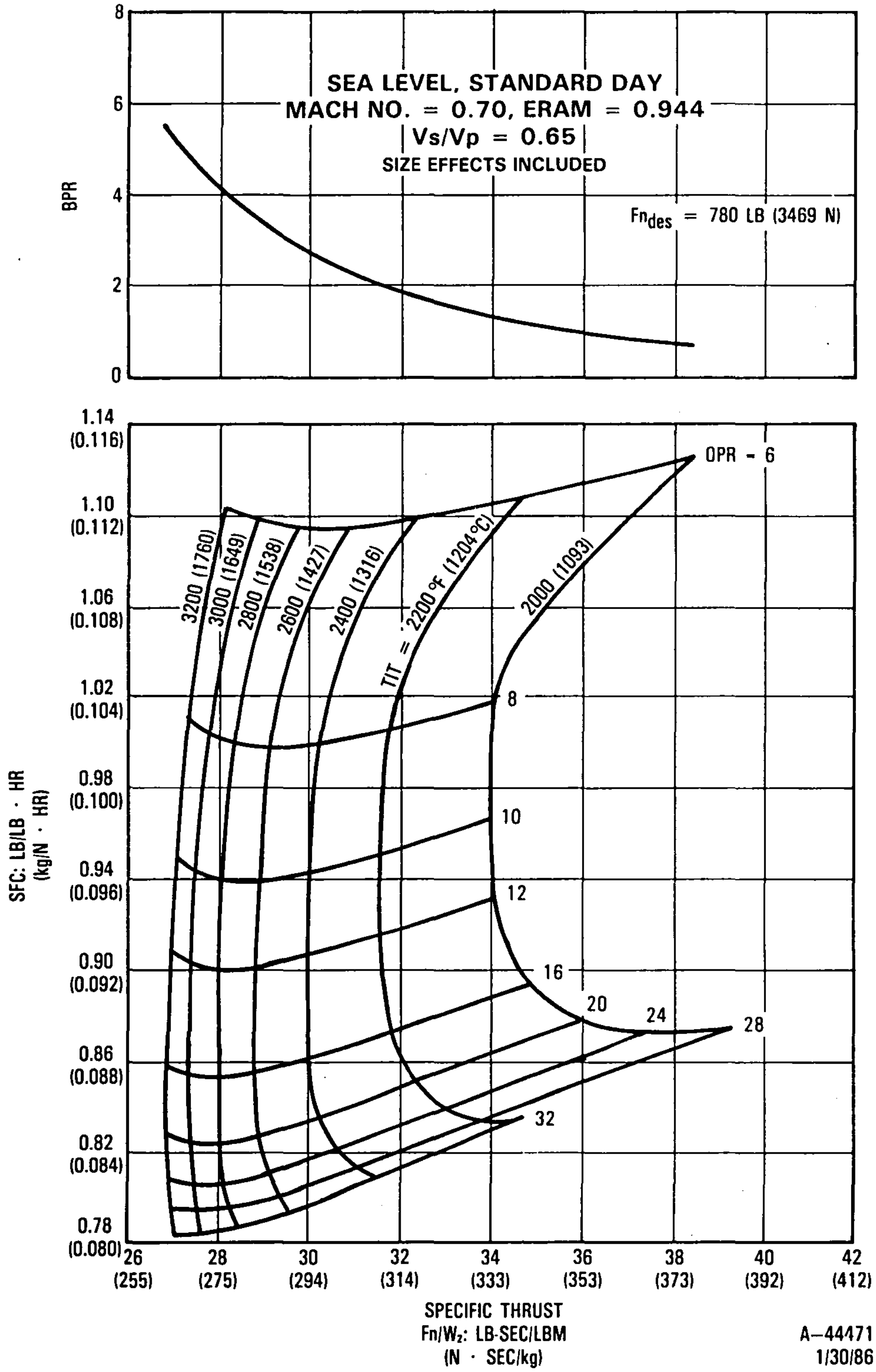


Figure 15. Advanced Turbofan Engine (Fan PR = 2.3).

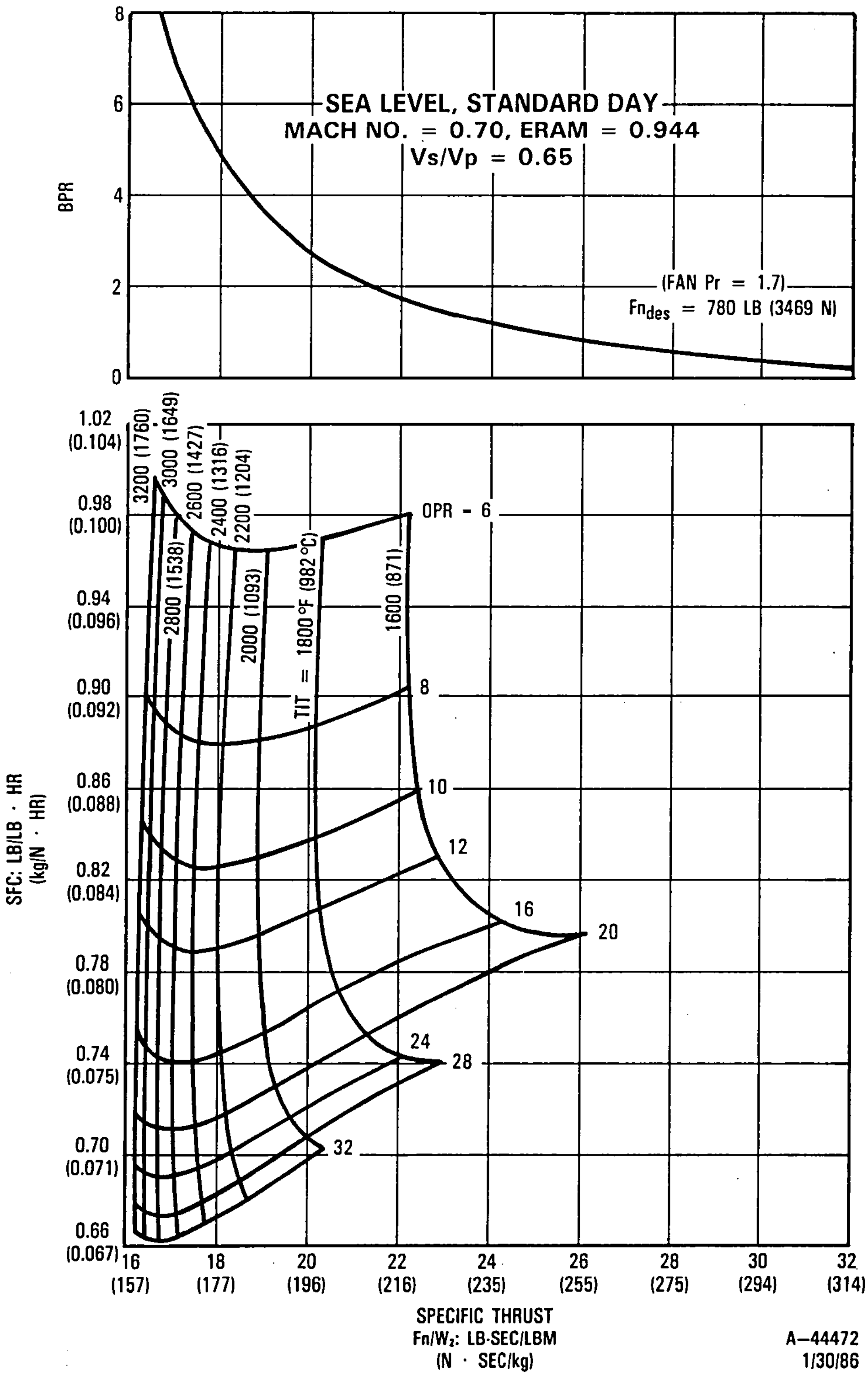


Figure 16. Advanced Turbofan Engine - Size Effects Not Included.

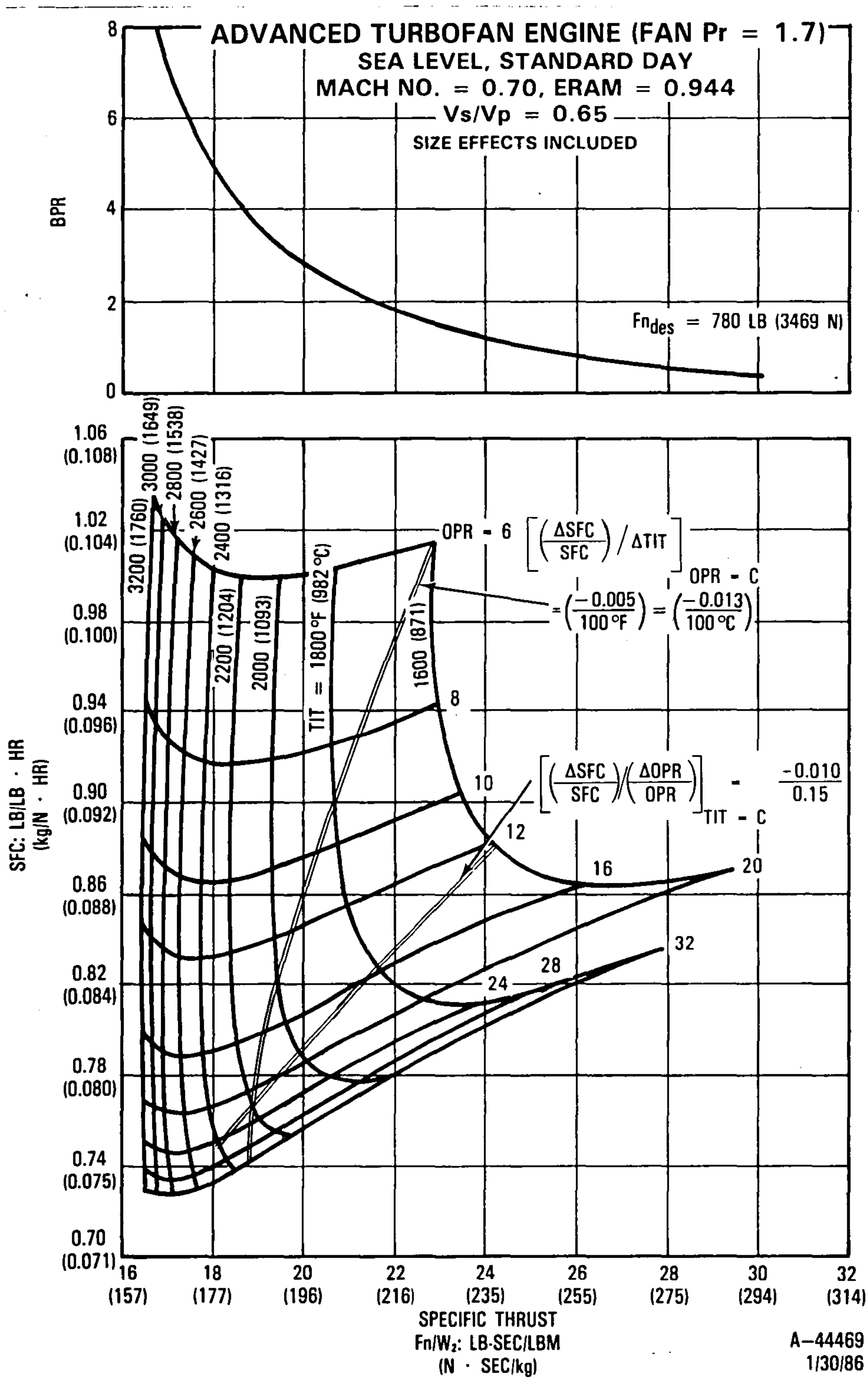


Figure 17. Example of Superimposed Improvement Derivatives.

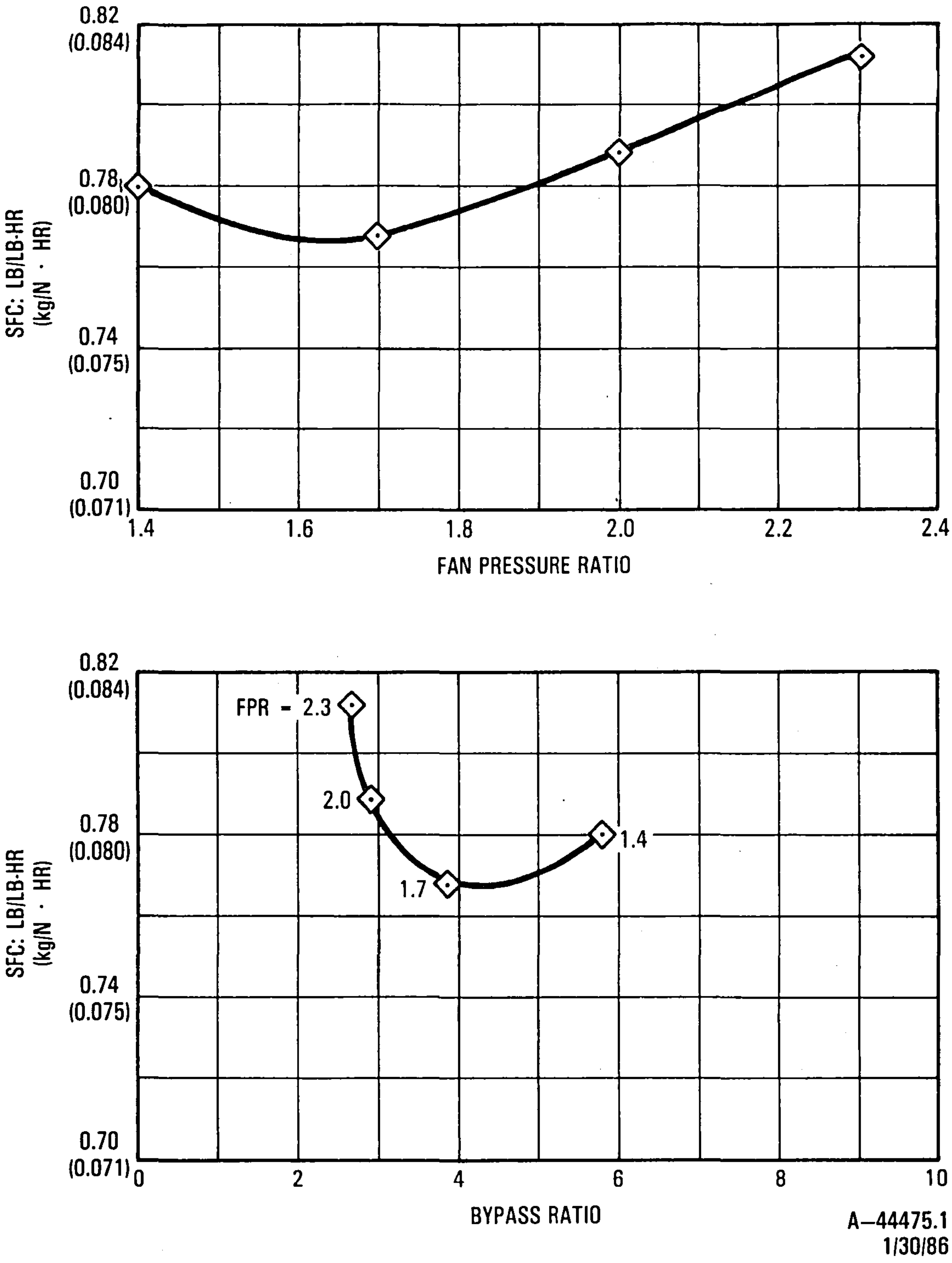


Figure 18. Effect of Fan Pressure Ratio - Advanced Turbofan.

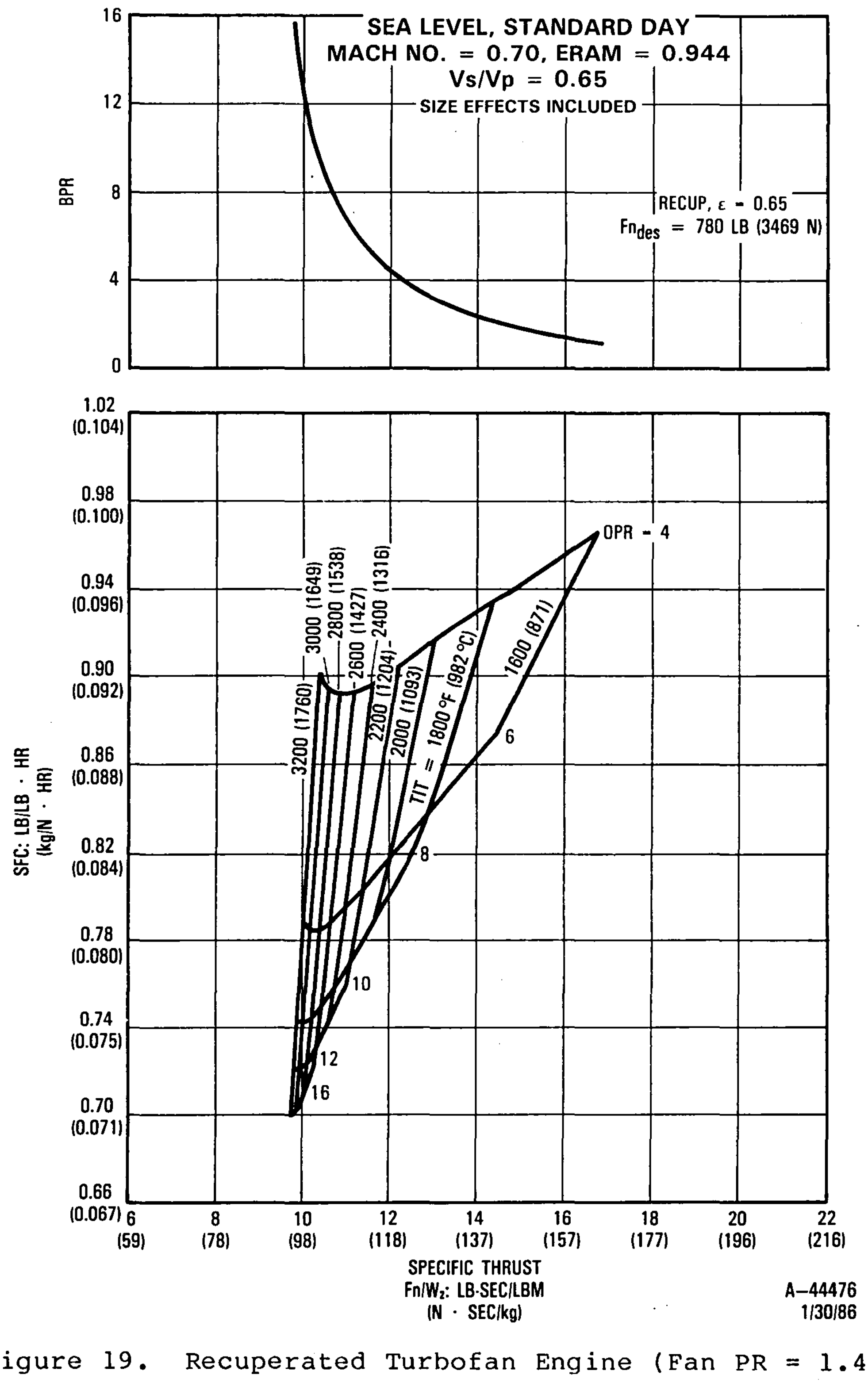
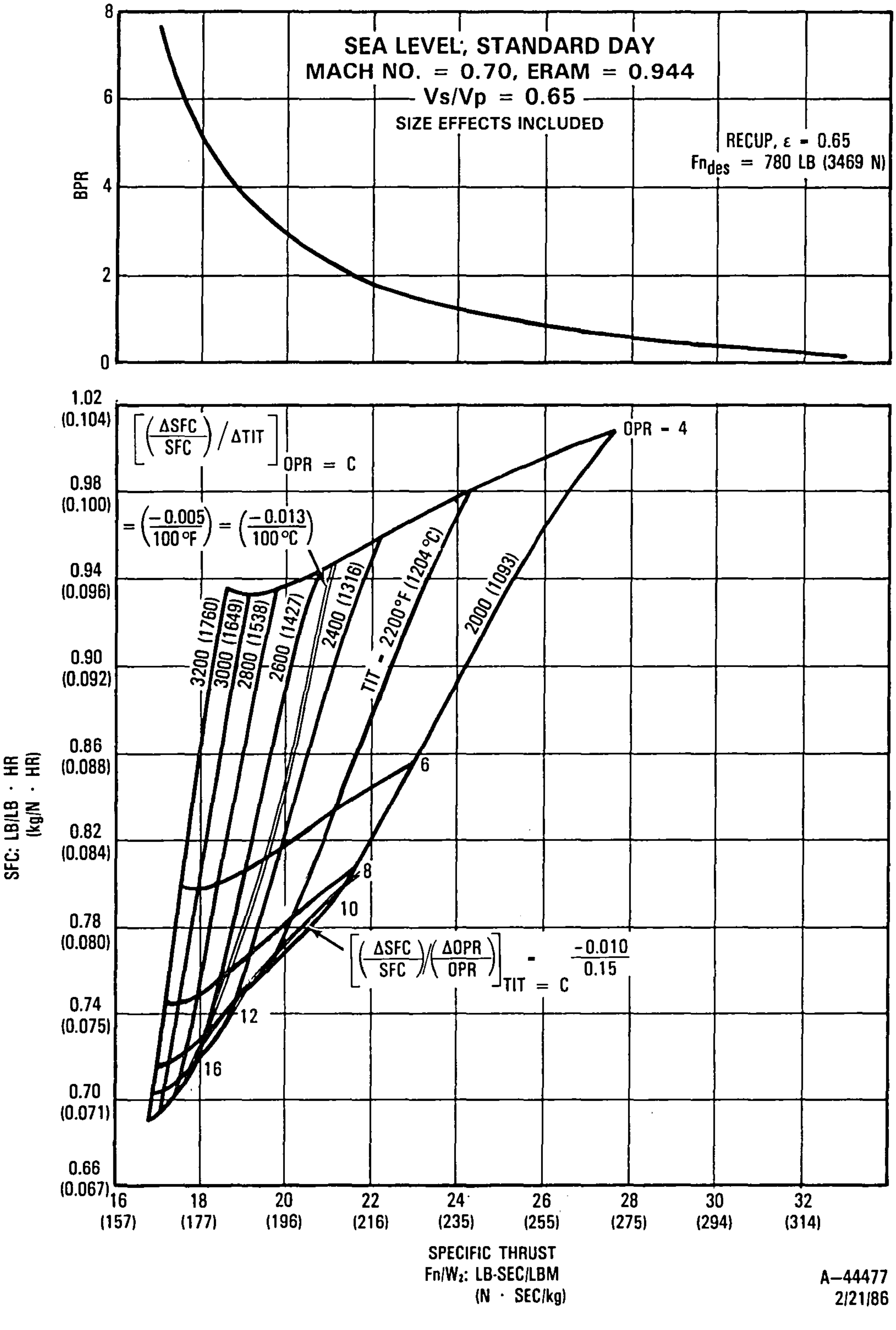


Figure 19. Recuperated Turbofan Engine (Fan PR = 1.4).



$$\left[\left(\frac{\Delta SFC}{SFC}\right)/\Delta TIT\right]_{OPR\,=\,C} = \left(\frac{-0.005}{100\,^{\circ}F}\right) = \left(\frac{-0.013}{100\,^{\circ}C}\right)$$

$$\left[\left(\frac{\Delta SFC}{SFC}\right)\left(\frac{\Delta OPR}{OPR}\right)\right]_{TIT\,=\,C} = \frac{-0.010}{0.15}$$

Figure 20. Recuperated Turbofan Engine (Fan PR = 1.7).

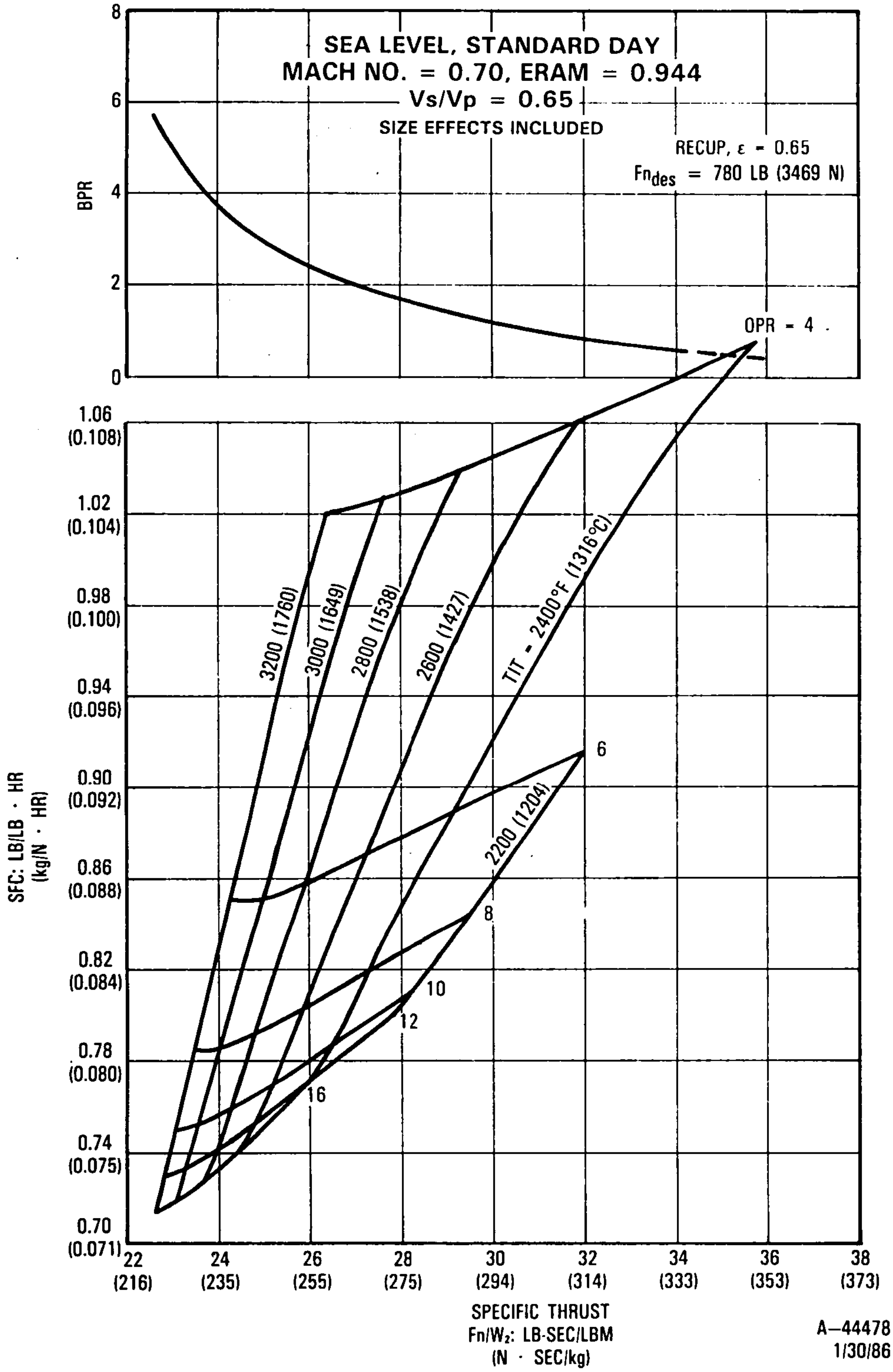


Figure 21. Recuperated Turbofan Engine (Fan PR = 2.0).

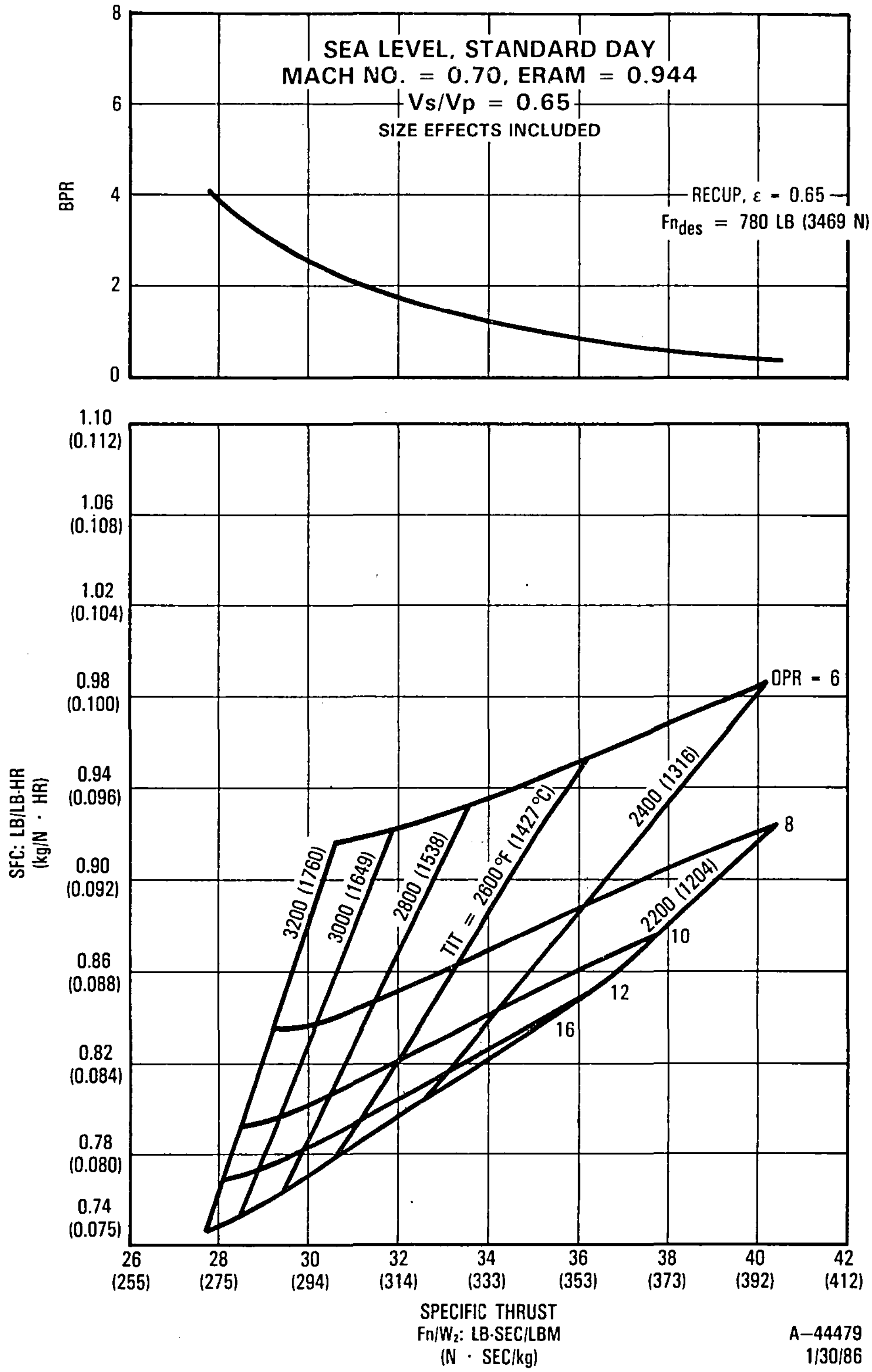


Figure 22. Recuperated Turbofan Engine (Fan PR = 2.3).

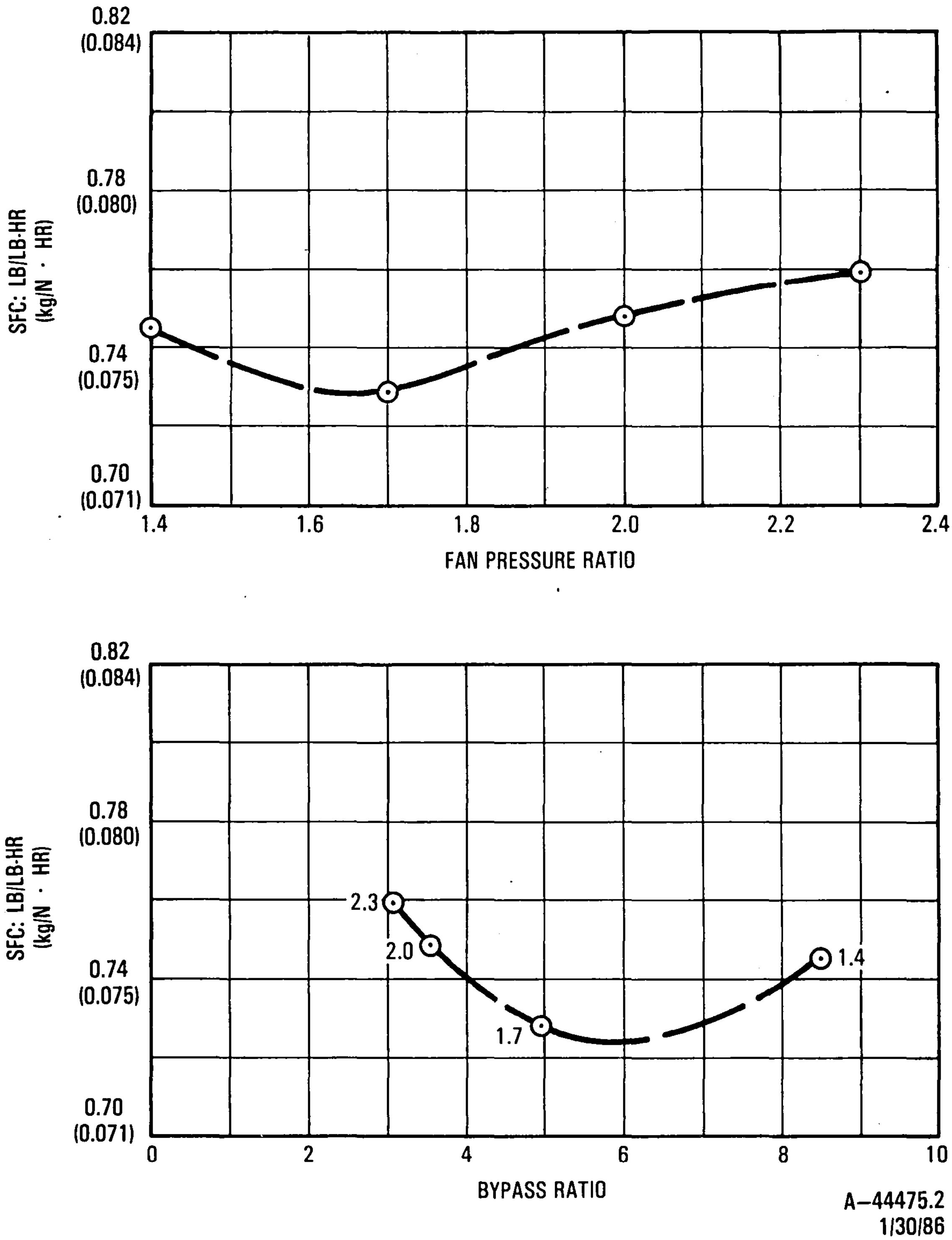


Figure 23. Effect of Fan Pressure Ratio - Recuperated Turbofan (Recuperator Effectiveness = 0.65).

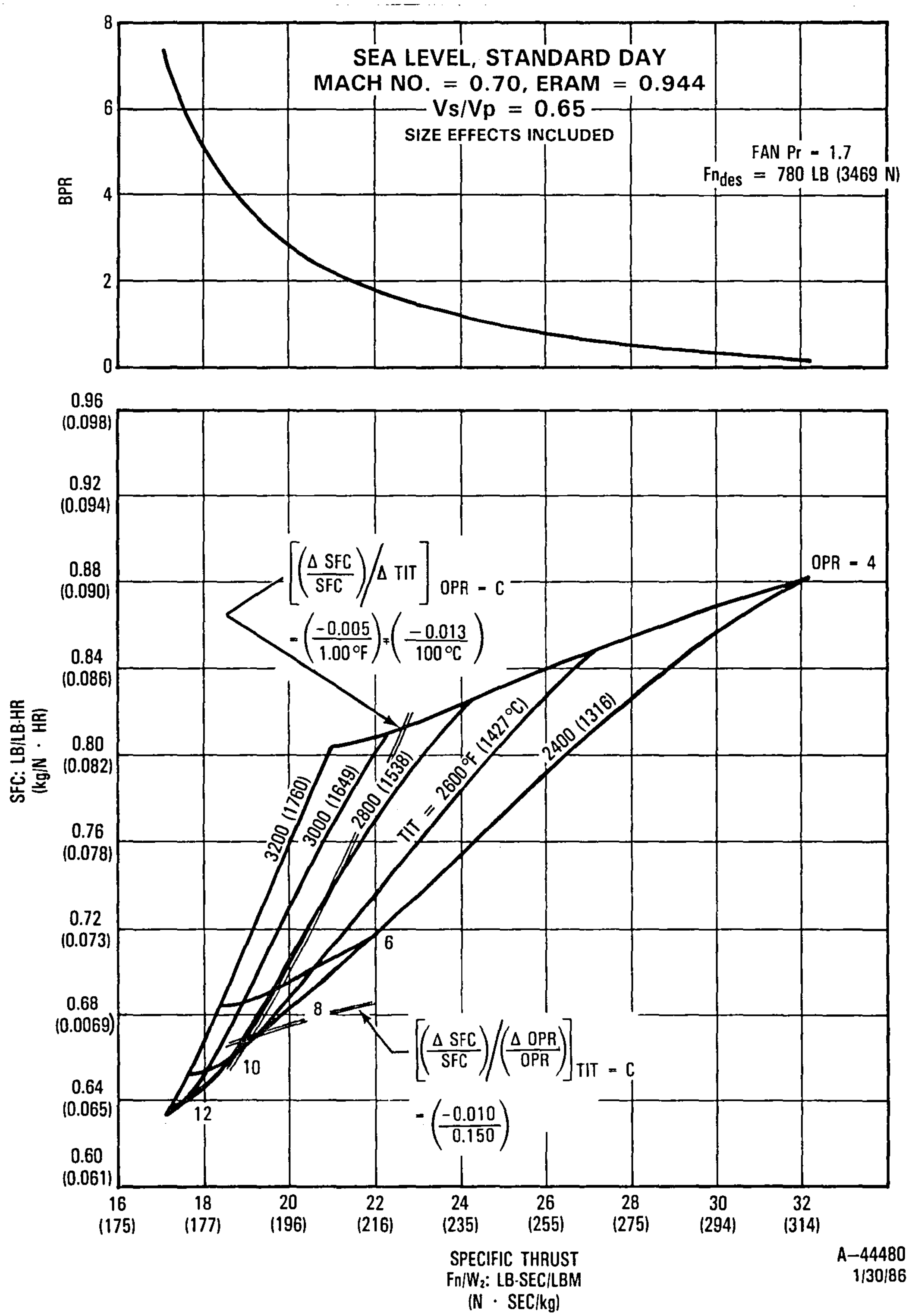


Figure 24. Recuperated Turbofan Engine (Recuperator Effectiveness = 0.85).

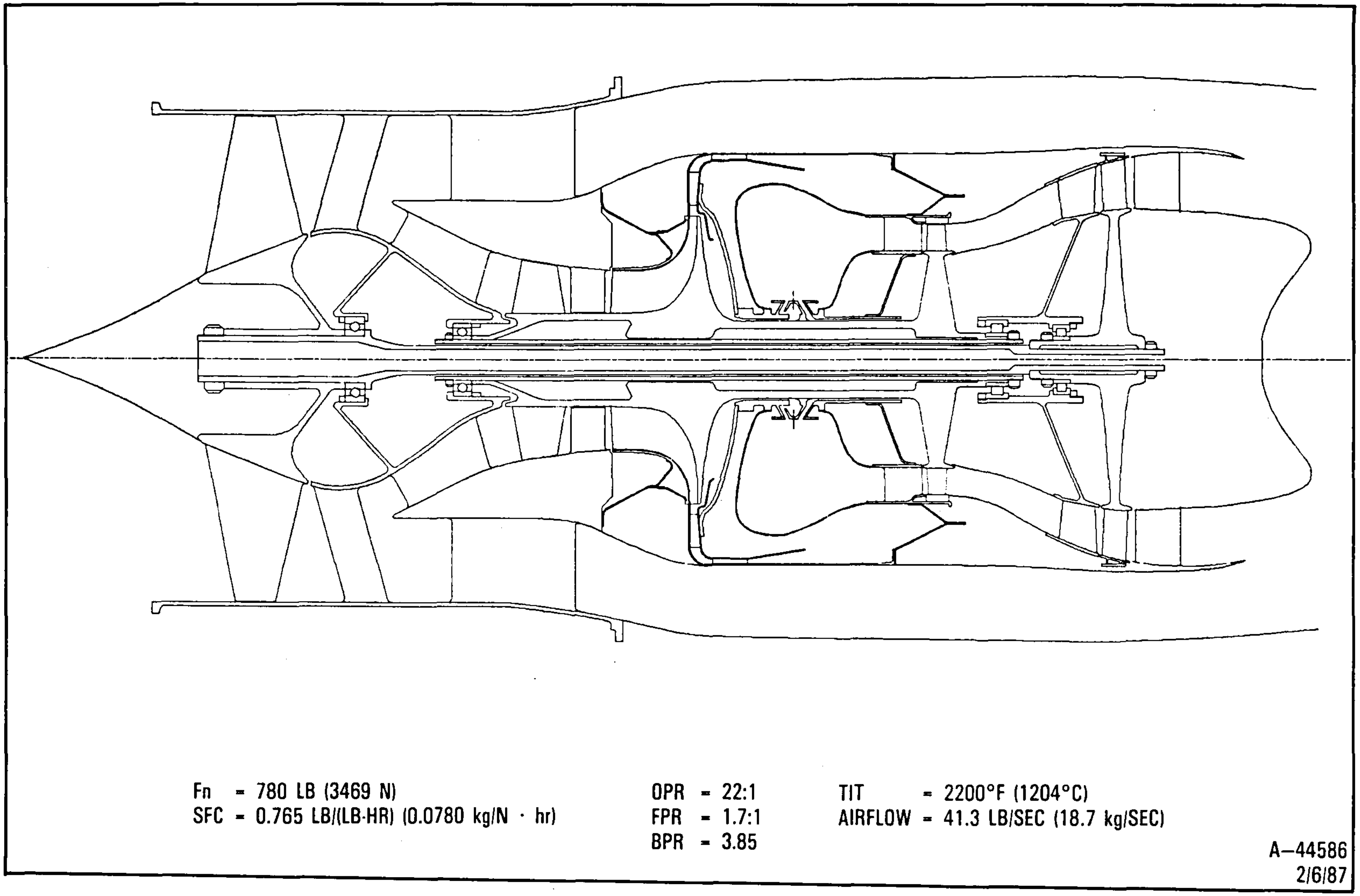


Figure 25. Advanced Turbofan Engine.

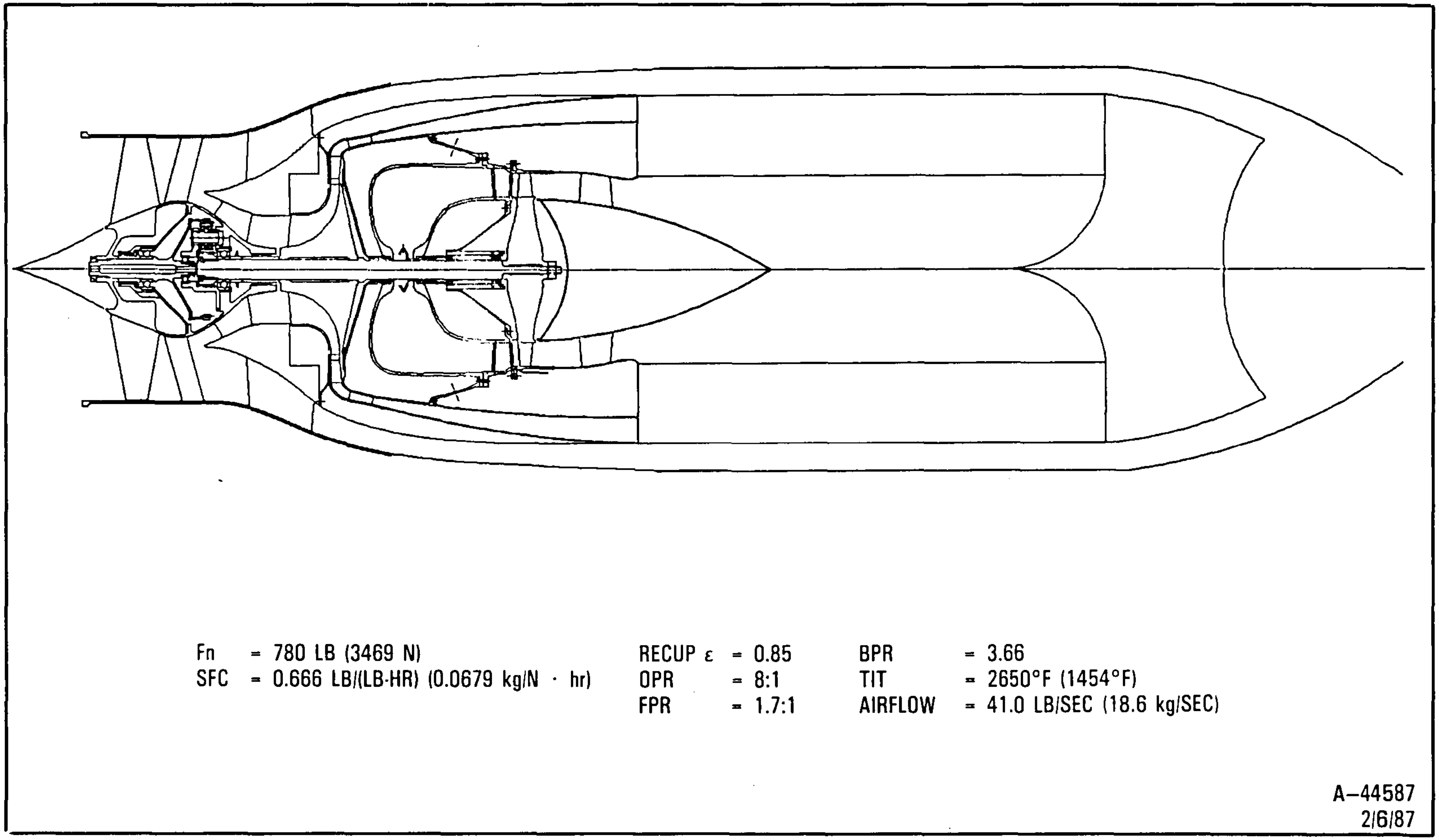


Figure 26. Recuperated Turbofan Engine.

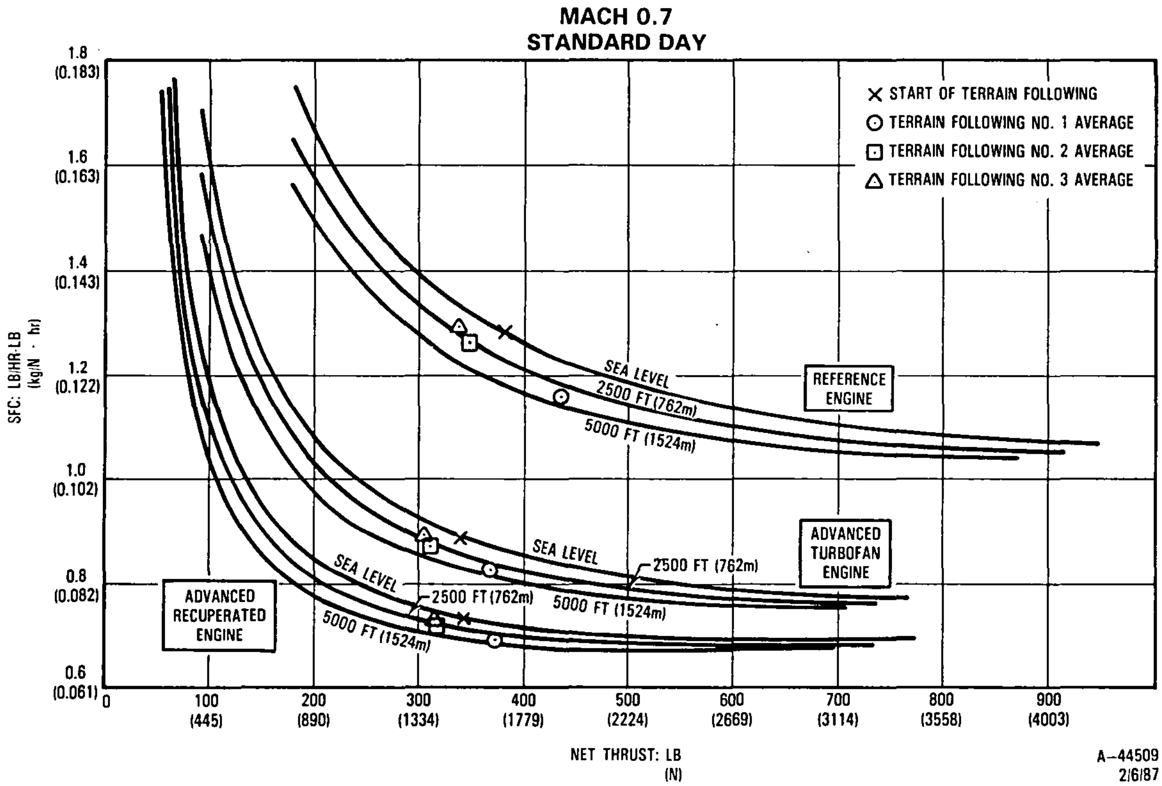


Figure 27. Engine Operating Points - Terrain Following.

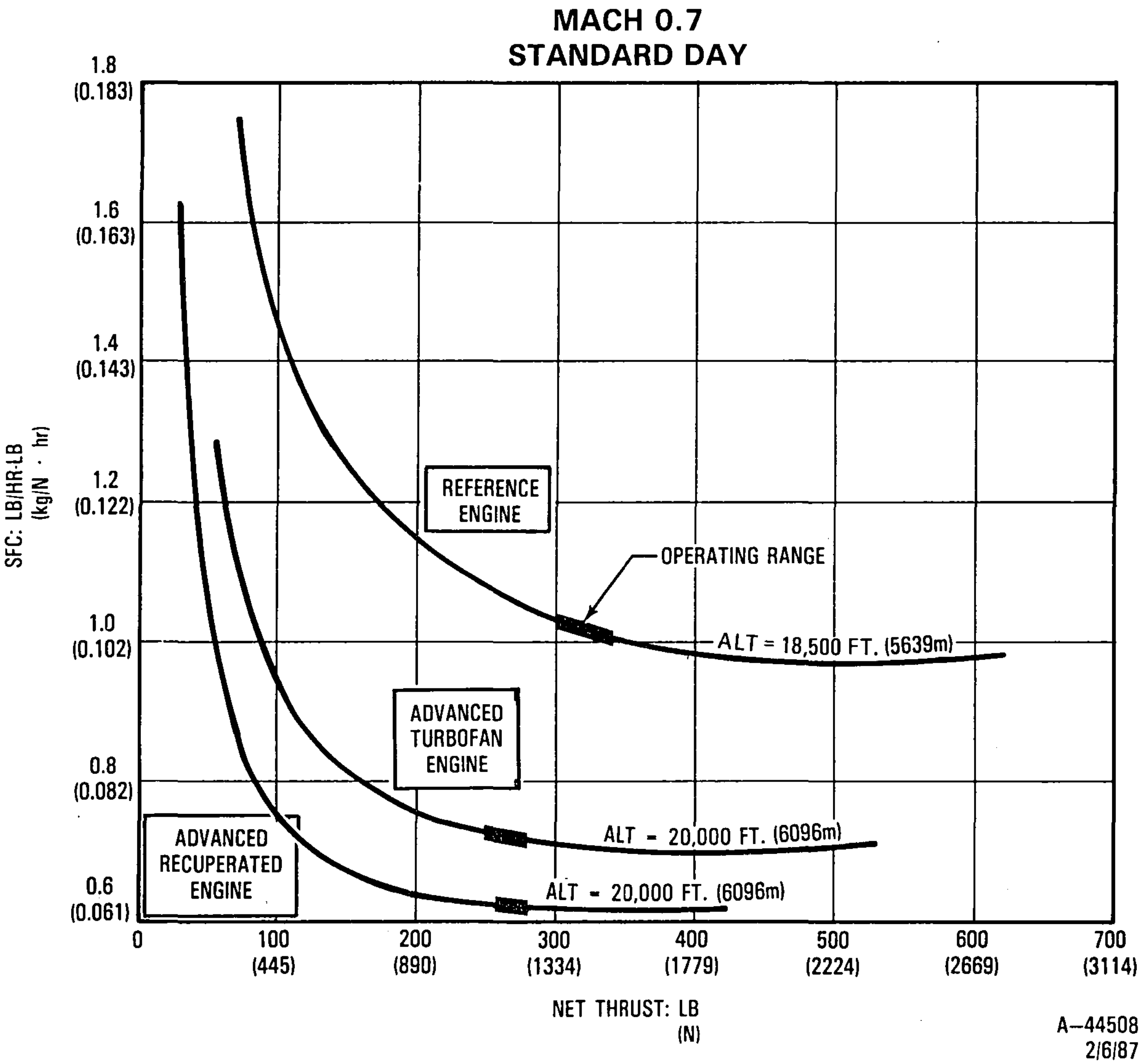


Figure 28. Optimum Altitude Cruise Engine Operating Range.

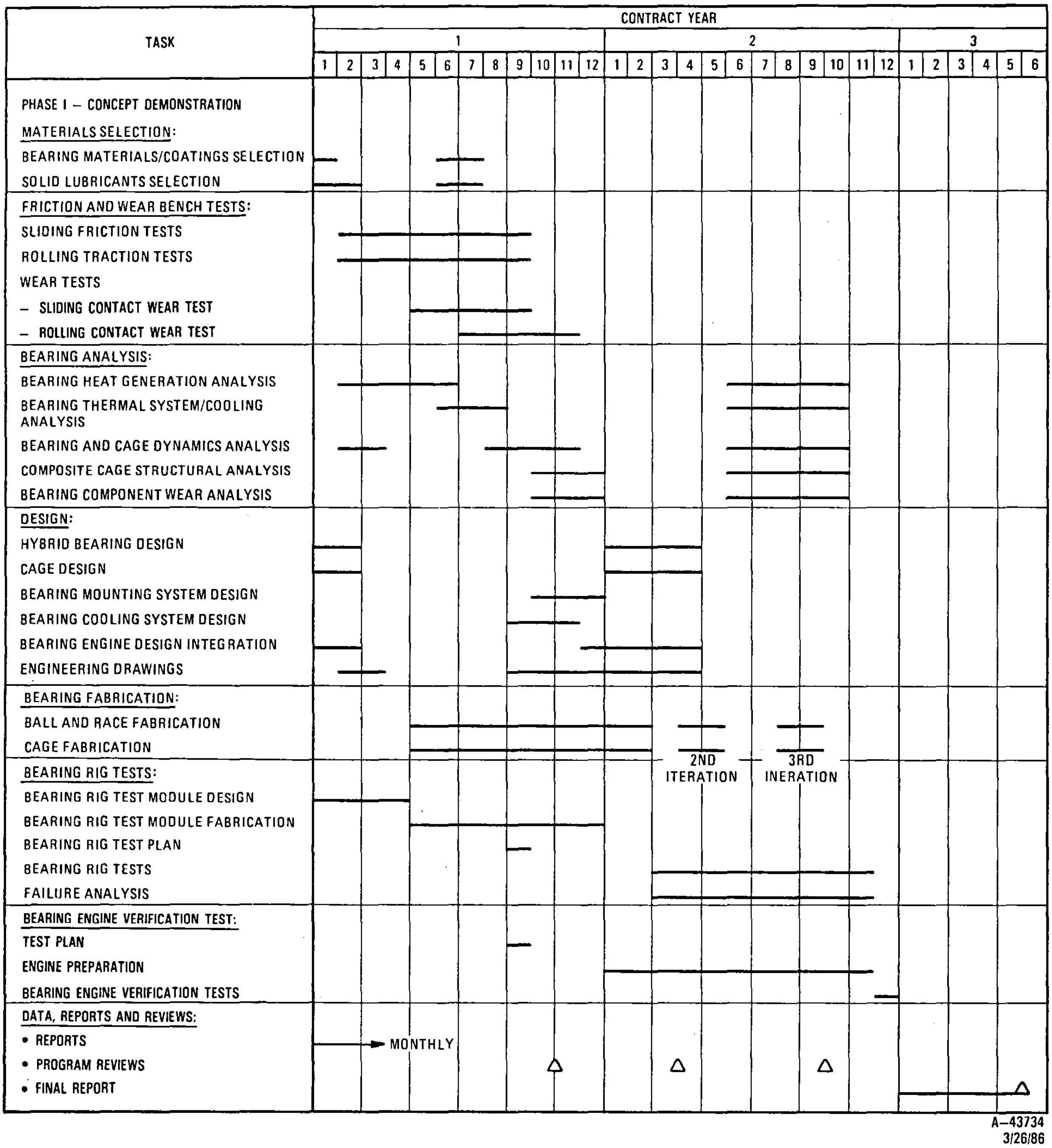


Figure 29. Solid-Lubricated Bearings, Phase I Program Schedule.

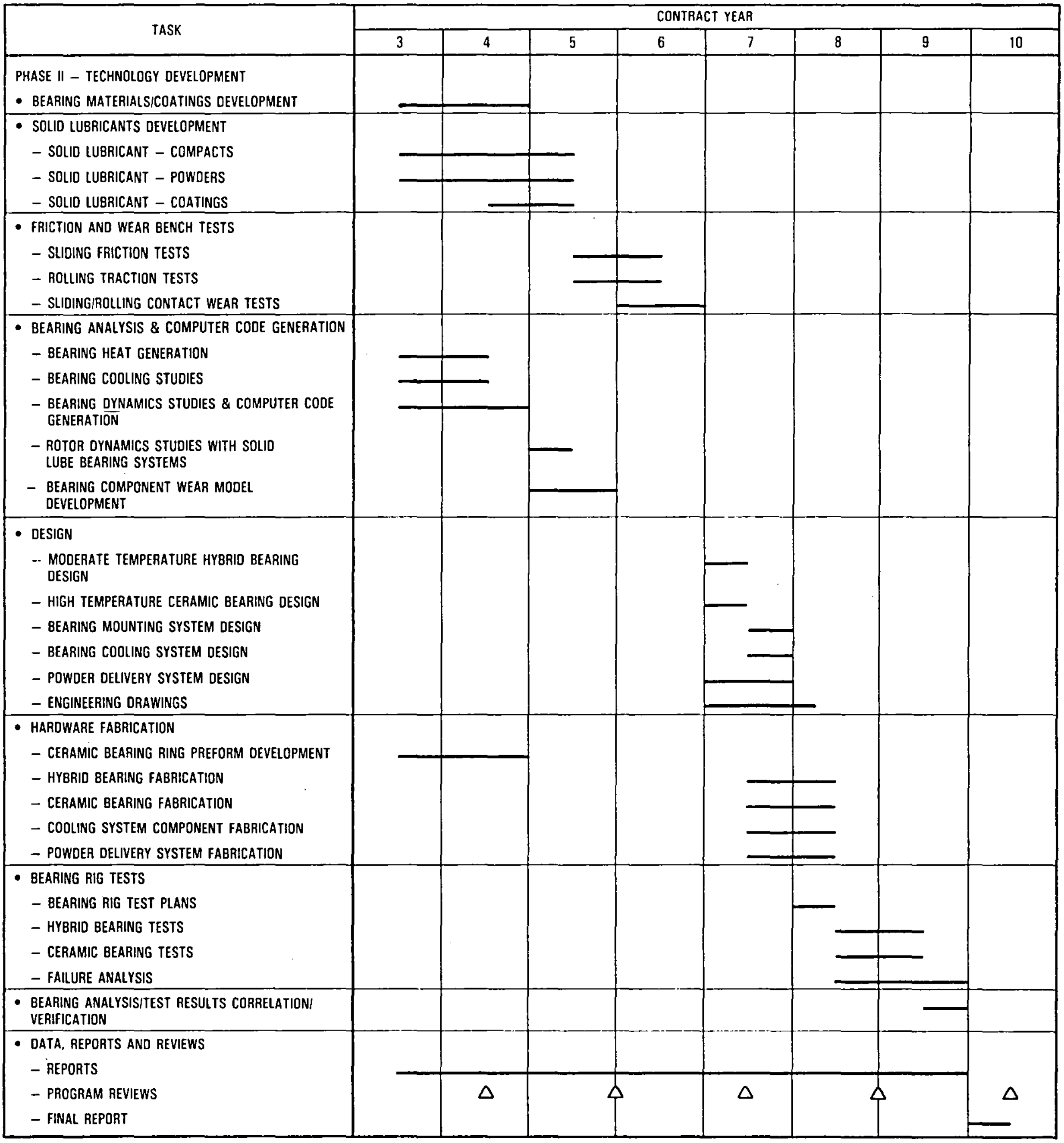


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Figure 30. Solid-Lubricated Bearings, Phase II Program Schedule.

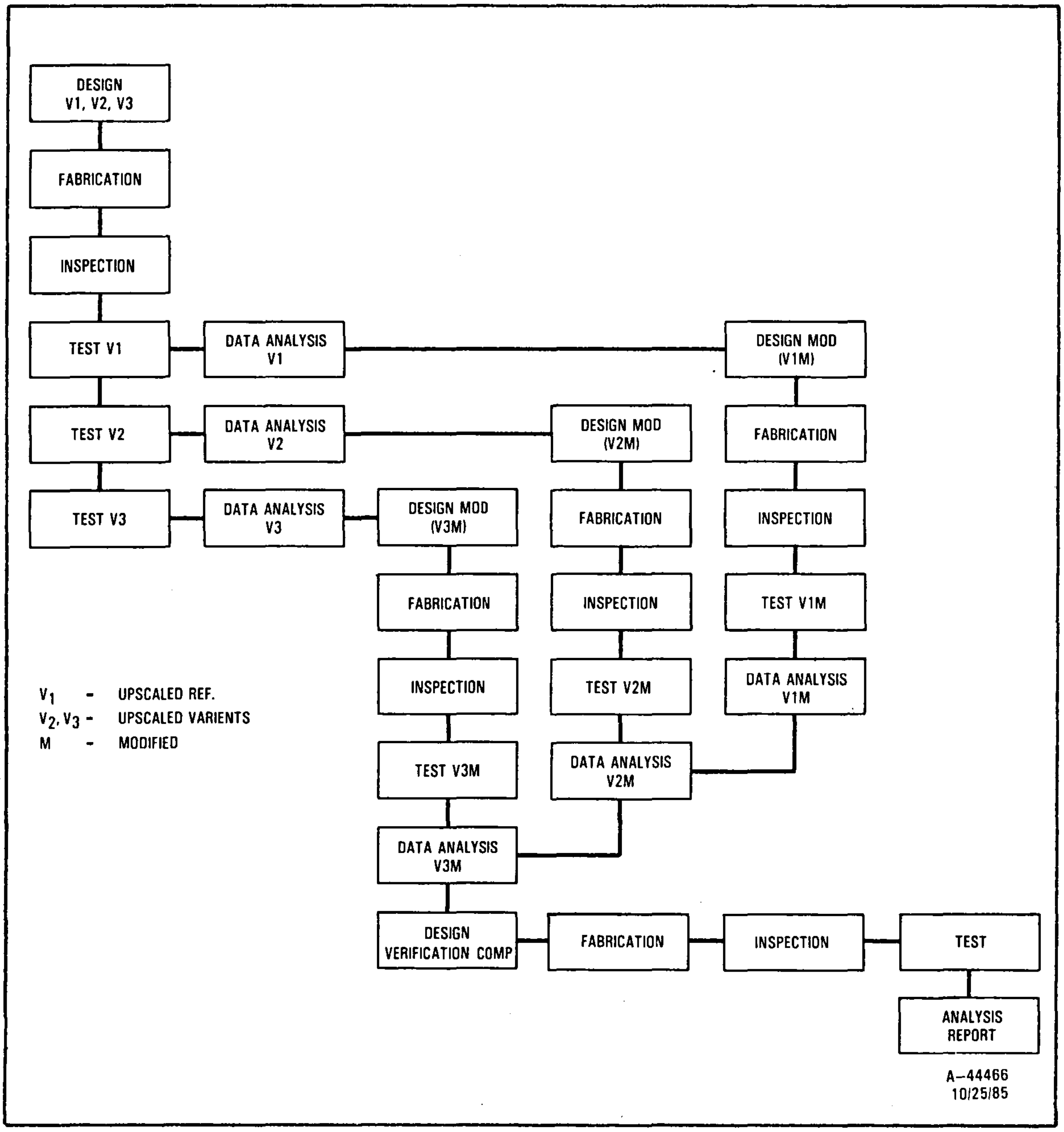

Figure 31. Small Component Program Test Sequence.

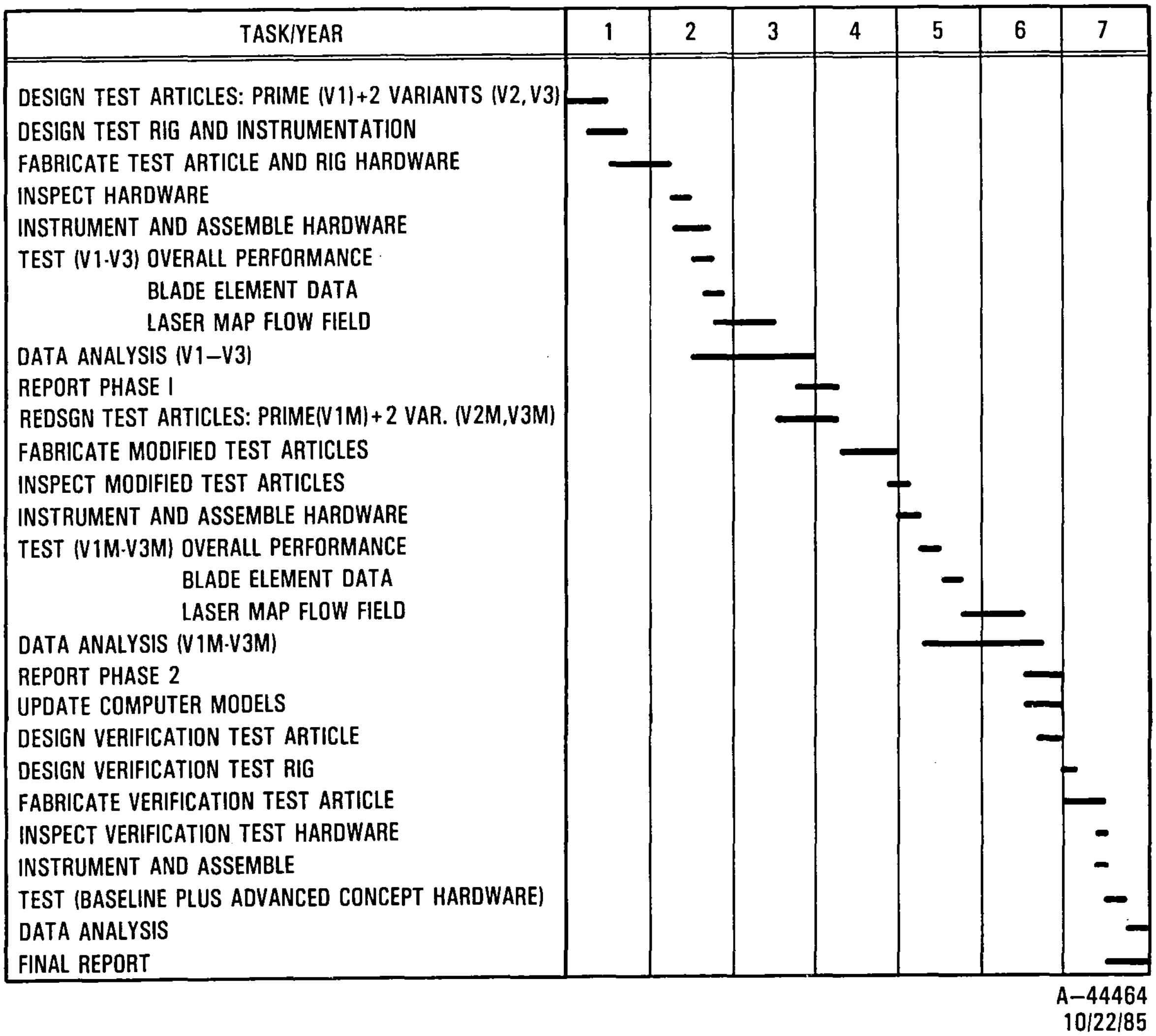

Figure 32. Small Component Program Test Schedule.

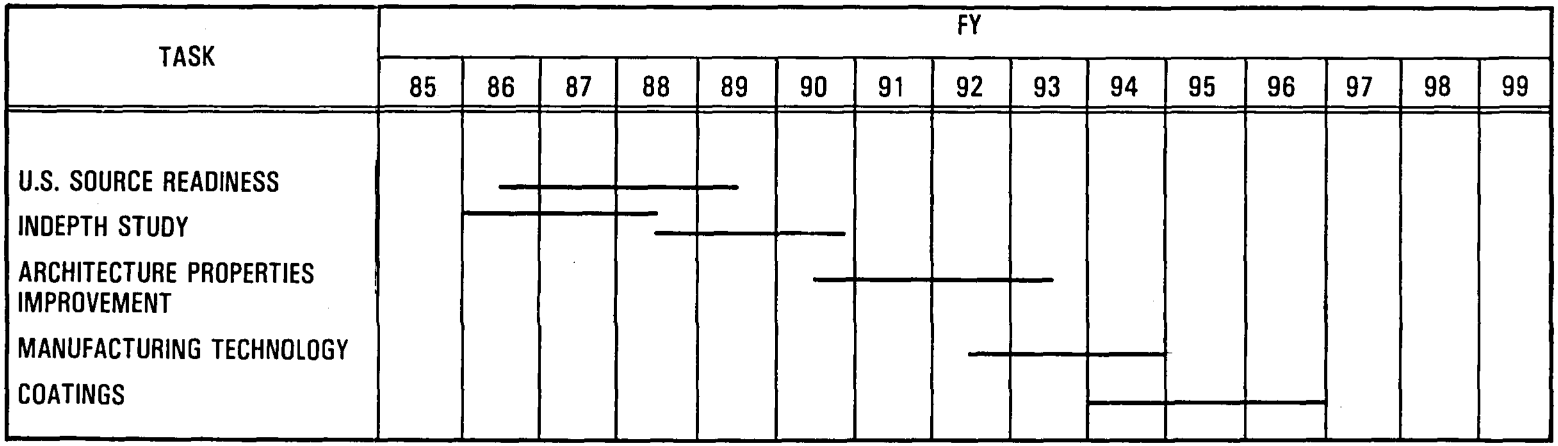

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Figure 33. Ceramic Composite Materials Program Schedule.

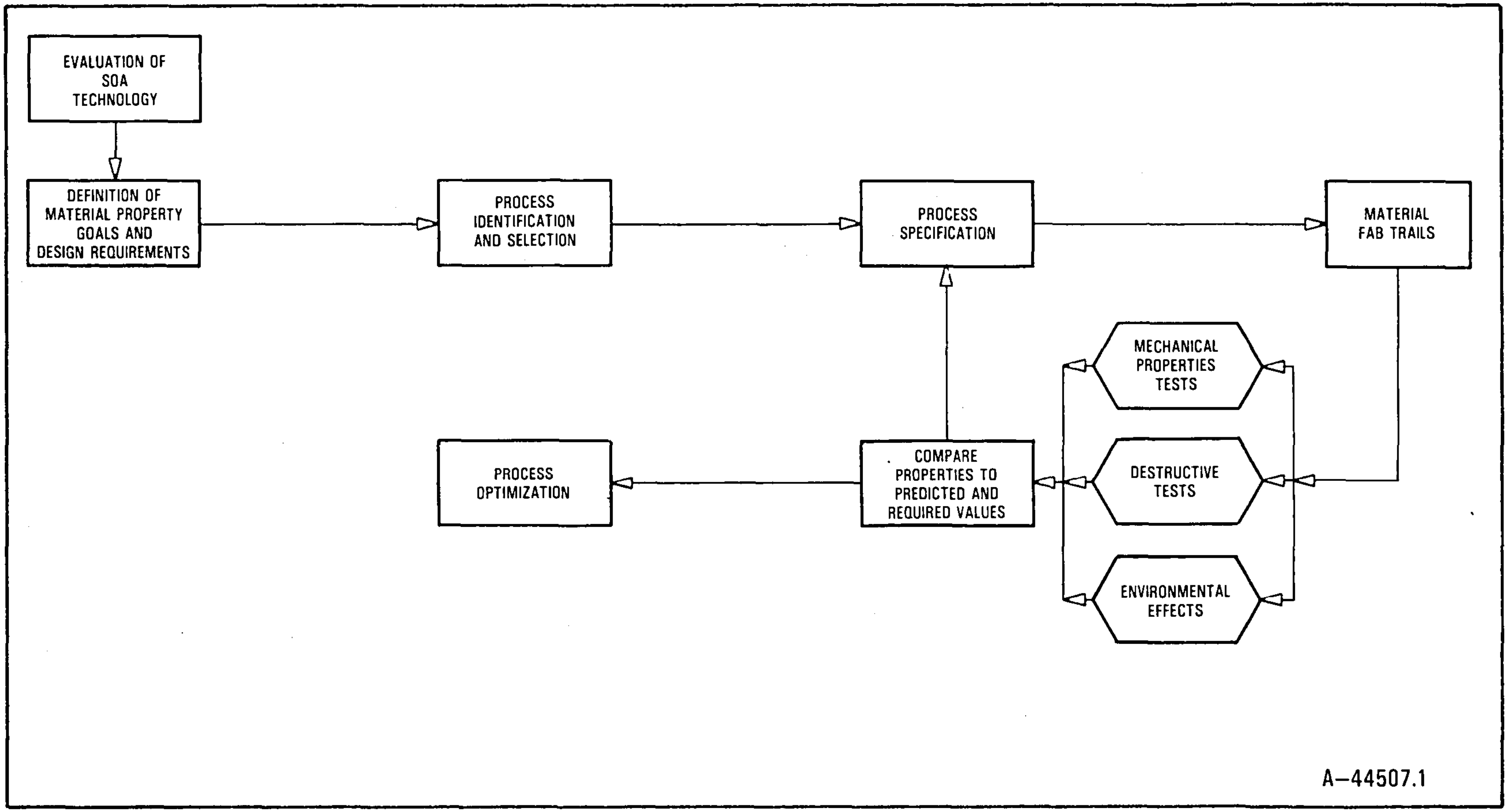

Figure 34. Task I: Domestic Source Readiness Roadmap.

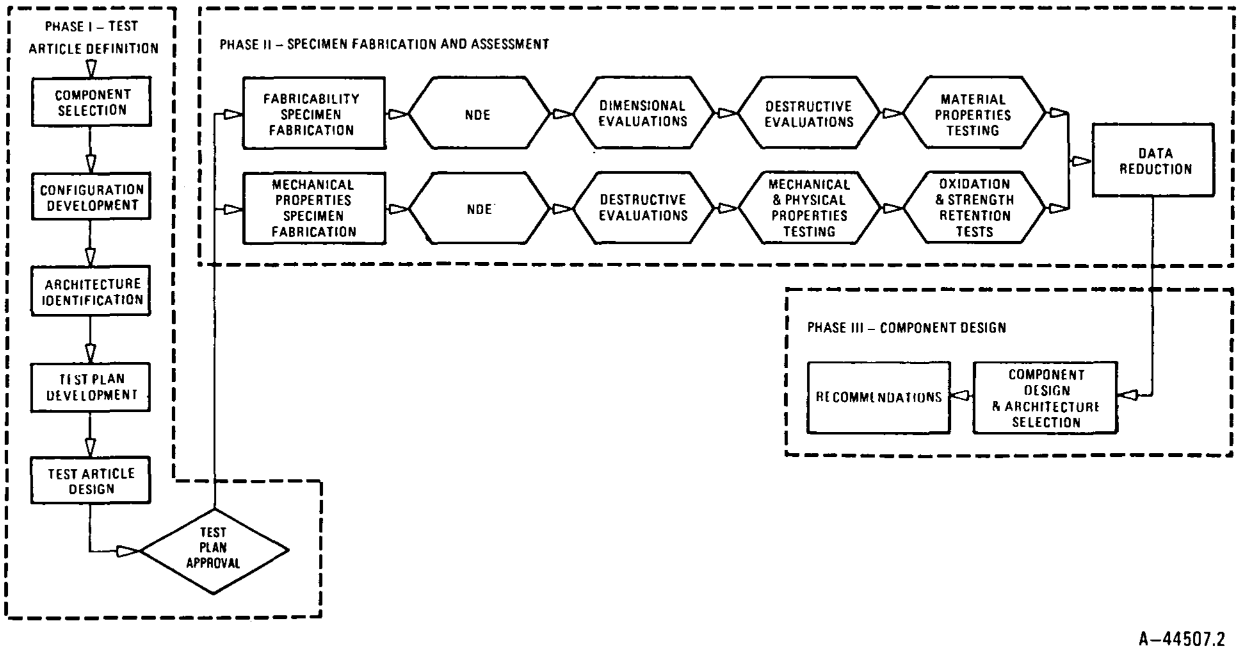

Figure 35. Task II: In-Depth Component Feasibility Study Roadmap.

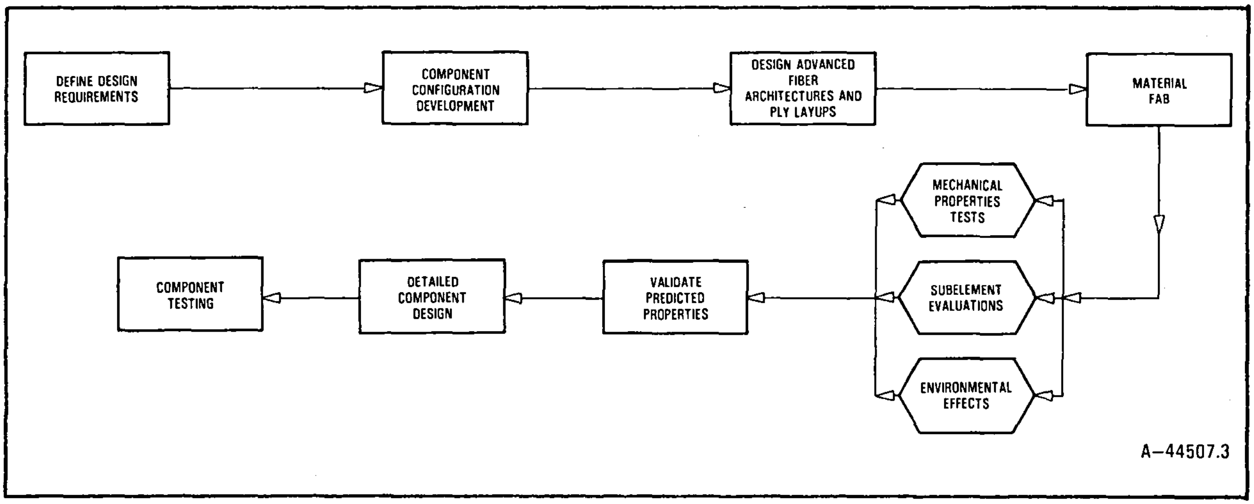

Figure 36. Task III: Advanced Architecture Property Optimization Roadmap.

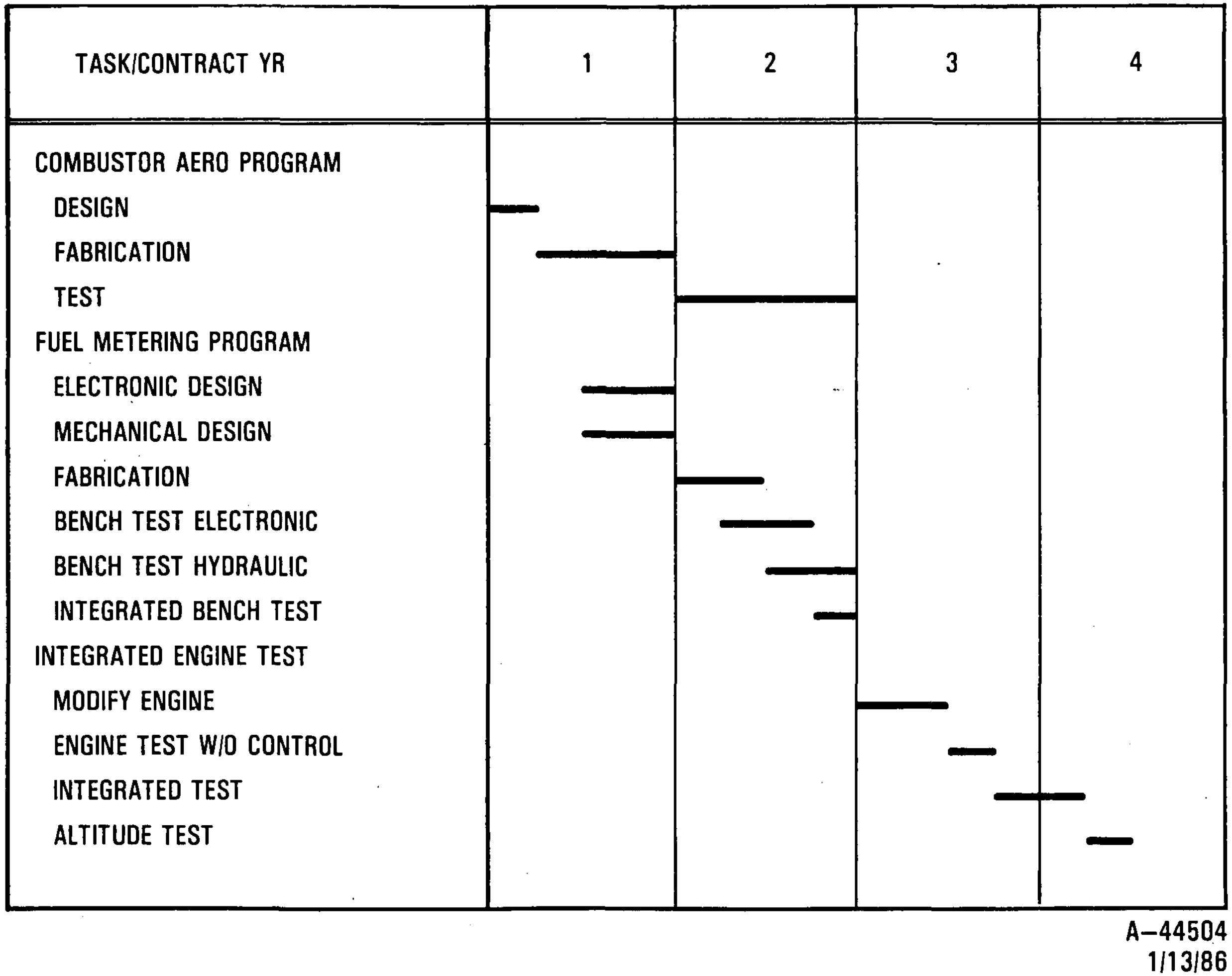

Figure 37. Carbon Slurry Technology Program Schedule.

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6. "The Cruise Missile Era Dawns", Interavia, 9 1982.

APPENDIX A

LIST OF SYMBOLS AND ABBREVIATIONS

ADV	advanced
AFWAL	Air Force Wright Aeronautical Laboratories
AGL	above ground level
ALCM	air-launched cruise missile
ASD	Advanced System Development, Wright Patterson AFB
AVG	average
BPR	bypass ratio
BRG	bearing
Btu	British thermal unit
C	Constant, carbon
CAT	Computer-Aided Tomography
CD	drag coefficient
CL	lift coefficient
CVD	chemical vapor deposited
DARPA	Defense Advanced Research Projects Agency
ERAM	inlet ram efficiency
FAB	fabrication
FN	net thrust
FPR	fan pressure ratio
GLCM	ground-launched cruise missile
HP	high pressure
IP	intermediate pressure
ISMAP	Integrated Systems Mission Analysis Program
K	Kelvin
LCC	life cycle cost
LP	low pressure
Lengine	engine length
Lfuel	effective fuel tank length
Lhard	effective hardware length
M	Mach No.
MFG	manufacturing
N	Newton
N.M.	nautical miles
N.MI	nautical miles
N.MILES	nautical miles
N1	fan shaft speed (RPM)
N2	gas generator speed (RPM)
N/A	not applicable
NASA	National Aeronautics and Space Adminstration
NDE	non-destructive evaluation
ODS	oxide dispersion strengthened
OPR	overall pressure ratio
PR	pressure ratio
PT	total pressure
PUC	production unit cost (percent of reference engine)

PWR	power
R	Rankine
rpm	revolutions per minute
SECT	Small Engine Component Technology
SEM	scanning electron microscopy
SEP	Société Européènne de Propulsion in France
SFC	specific fuel consumption
SLCM	sea-launched cruise missile
SiC	silicon carbide
SiN	silicon nitride
TIT	turbine inlet temperature
TRIT	turbine rotor inlet temperature
U.S.	United States
V1	Upscaled reference
V1M	modified upscaled reference
V2	upscaled varient
V2M	modified upscaled varient
V3	upscaled varient
V3M	modified upscaled varient
VAR	varient
VP	velocity of the core gas at the mixing plane
VS	velocity of the bypass air at the mixing plane
W	airflow rate
WA	airflow rate
WAF	airframe weight
WFIX	fixed weight
WFUEL	fuel weight
WGROSS	gross weight
WPROP	propulsion system weight
Wt	weight
cm	centimeters
ft	feet
hr	hours
in	inches
kJ	kilojoule
kN	kilonewton
kg	kilogram
ksi	1000 pounds per square inch
lb	pounds
lbm	pounds mass
m	meter
max	maximum
min	minimum
mrad	milliradian
rad	radian
s	second
sec	second
spec	specification
°C	degrees centigrade
°F	degrees Fahnenheit

δ	(total pressure)/(standard pressure)
Δ	difference in
η	efficiency
θ	(total temperature)/(standard temperature)
ξ	effectiveness
π	(circumference/diameter) circle
ω	airflow rate

Subscripts

0	freestream
2	at the compressor face
des	design

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