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# STUDY OF TURBOFAN ENGINES DESIGNED FOR LOW ENERGY CONSUMPTION

## FINAL REPORT

by

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UNITED TECHNOLOGIES CORPORATION

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16. Abstract

Declining U.S. oil reserves and escalating energy costs underline the need for reducing fuel consumption in aircraft engines. This report identifies near-term technology improvements which can reduce the fuel consumed in the JT9D, JT8D, and JT3D turbofans in commercial fleet operation through the 1980's. Projected technology advances are identified and evaluated for new turbofans to be developed after 1985. Programs are recommended for developing the necessary technology.

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## FOREWORD

The work described herein, which was conducted by Pratt & Whitney Aircraft Division of United Technologies Corporation, was performed under NASA Project Manager, Mr. James Dugan, of NASA Lewis Research Center. The report was prepared by D. E. Gray, the Pratt & Whitney Aircraft Program Manager, assisted by W. O. Gaffin, F. D. Havens, F. D. Streicher, and R. A. Lewis, Jr.

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# 1.0 SUMMARY

Improvements which have the potential for reducing fuel consumption in current and future subsonic transport turbofan engines were evaluated in this NASA-sponsored program. With these improvements, fuel consumption could be reduced by 3 to 7 percent in the current commercial fleet, and by 16 to 17 percent in future subsonic transport aircraft.

The P&WA JT9D-7, the JT8D-9, and the JT3D-3B engines were used for evaluating improvements to current turbofan engines. These engines were selected because they are representative of the turbofan powerplants now in commercial service.

The evaluation of improvements for future turbofan engines was based on engine technology required for a 1985 engine development start date. Parametric cycle evaluation and a more detailed analysis of several promising cycles led to the definition of a representative future fuel conservative turbofan.

The Boeing 747, 727, and 707 aircraft were used in the current engine evaluation to relate the projected fuel savings to actual airline operation. A three-engine medium range and a four-engine long range aircraft were used for the evaluation of future turbofan engines. The advanced aircraft definitions were provided by the NASA-Lewis, Langley, and Ames Research Centers.

The work under this contract was concluded with the formulation of various programs designed to acquire the technology necessary to achieve improvements found possible in this study.

Fuel consumption and airline direct operating cost (DOC) were used as figures-of-merit in the evaluation to define critical technology requirements. Fuel consumption calculations were based on representative flight stage lengths and aircraft utilization. The DOC includes the effects of investment, maintenance, and fuel costs associated with the engine design improvements. These costs, directly incurred by the airlines' operation, are an important measure of the economic benefits associated with energy savings.

In the long term, fuel consumption savings will almost certainly have to be achieved with increasingly more stringent pollution control requirements. Determining the effect of advanced fuel conservative technology on engine noise and exhaust emissions was an integral part of the work conducted during this program.

The results of the program are summarized in the following paragraphs.

Lowering Fuel Consumption

Lowering fleet fuel consumption by 3 to 7 percent is possible in current commercial turbofan engines. These savings can be achieved by internal and installation modifications, each one of which has the potential for reducing fuel consumption by at least 0.4 percent. The most direct means of reaching these levels are by (1) selective reblading in the fan and compressor and by improving the flowpath sealing and (2), the addition of an exhaust mixer installation to the separate exhaust stream JT9D and JT3D engines.

A potential for lowering fuel consumption by 6.5 percent was calculated for the JT9D-7 engine. This reduction is divided approximately equally between internal engine improvements and the use of an exhaust mixer. The potential savings for the JT8D and JT3D engines are 3.3 percent and 7.3 percent, respectively. The potential savings for the JT8D reflects a smaller benefit for forceably mixing the existing common flow engine exhaust.

Reducing future commercial fleet fuel consumption by over 15 percent was found possible with 1985 fuel conservative turbofan technology. Approximately two-thirds of this potential is associated with aerodynamic improvements, advanced materials, reduced turbine cooling, and reduced blade tip clearance in the high spool portion of the engine. Cycle pressure ratios in excess of 40:1 were found to be desirable when considering the full potential to be offered by these high spool technology improvements. The remaining third of the fuel savings potential consists of improved clearance control and improved aerodynamics in the low spool components, and the use of lightweight, high strength materials. A turbofan which embodies these advances was defined and given the Pratt & Whitney Aircraft study designation STF 477. The major cruise design cycle parameters of this engine are: a cycle pressure ratio of 45:1, a fan pressure ratio of 1.7, a bypass ratio of 8.0, and a maximum combustor exit temperature of 1427°C (2600°F).

Cursory studies show that the separate exhaust streams with properly matched jet velocities can achieve fuel consumption levels comparable to those of a mixed exhaust installation. However, anticipated thrust growth versions of any engine will tend to mis-match the exhaust stream velocities. In this case, the addition of an exhaust mixer will have greater benefits in lowering fuel consumption, as shown by growth versions of current engines. Further study, based on high bypass ratio engine mixer tests, is recommended to determine more precisely the benefits of mixing in future turbofan engines.

## Improving Airline Economics

An engine DOC reduction of 1.4 percent was indicated for internally modified JT9D-7 engines. DOC reductions of 0.8 and 2.2 percent were calculated for internally modified versions of the JT8D-9 and JT3D-3B engines, respectively. These benefits decreased to very small values when considering retrofitting current engines.

Although future turbofans offered significant savings in fleet fuel costs, this benefit was largely offset by increased maintenance costs associated with the more sophisticated engines. These costs need not be appreciably higher than in current turbofan engines provided that the proper emphasis is placed on technology programs aimed at maintainability features of the engine.

## Impact of Fuel Conservation Technology On the Environment

Noise and exhaust emission levels were unaffected by internal modifications to current engines. The addition of a forced exhaust mixer could reduce engine jet noise.

The STF 477 advanced turbofan included advanced burner technology to improve emissions. Projected advances in burner technology could significantly reduce nitric oxide, unburned hydrocarbons, and carbon monoxide emissions to levels below those of current engines. Unburned hydrocarbon and carbon monoxide emissions are also below proposed Environmental Protection Agency (EPA) Standard levels. However, the attainment of the proposed EPA nitric oxide levels will require further technological advances.

Advanced technology will be required to reduce the noise generated by engine components and to increase the effectiveness of sound absorbing treatment to achieve the noise levels predicted for the STF 477 engine. It is projected that these advances in acoustic technology will provide a 4 to 5 EPNdB lower total noise compared to existing high bypass ratio engines. If achieved, this technology would permit the noise goal of 10 EPNdB below the present Federal Aviation Regulations Part 36 (FAR 36 minus 10 EPNdB) to be reached without the introduction of inlet rings or fan duct splitters.

## Recommended Technology Programs

Technology programs for reducing fuel consumption, and which are considered to be critical to current or advanced fuel conservative turbofan engines, were formulated. A total of 41 programs were recommended and a preliminary technical plan was established for each program. The programs address improvements in component aerodynamics, materials/cooling technology, structures/mechanics, and installation improvements. The programs generally consist of analytical design studies, component technology substantiation, and engine demonstration.

## 2.0 INTRODUCTION

The growing concern over diminishing fossil fuel supplies, compounded by escalating costs of oil-based energy, has stimulated the research and development of effective fuel conservation measures on a nationwide basis. As part of this effort, the National Aeronautics and Space Administration (NASA) has extended the scope of a continuing series of Advanced Transport Technology studies to include investigations directed towards minimizing fuel consumption in America's commercial aircraft fleet. These propulsion studies have encompassed the minimization of performance loss in current operational turbofan engines, as well as the study of fuel conserving turbofan and unconventional propulsion concepts which could be operational twenty years in the future.

A projection of fuel consumption by U.S. scheduled airlines to 1990 (Figure 2.0-1) illustrates the potential for fuel savings in subsonic transport turbofan engines. The JT8D and JT3D first generation turbofans currently consume over 80 percent of the total fuel or over 30 billion liters (8 billion gallons) annually in U.S. scheduled airline operation. In 1990, these two engines and the second generation JT9D turbofan are projected to use approximately the same amount of fuel annually. A one percent fuel consumption improvement in these engines alone could result in a 3.8-billion-liter (1-billion-gallon) fuel savings over a ten-year period. The JT9D engine, which is characterized by a higher pressure ratio, higher combustor exit temperature, and a higher bypass ratio fan to improve fuel economy, is expected to be a dominant commercial propulsion system into the 1990's. New engines, capable of offering substantially lower fuel consumption in the future, are also needed to meet the increasing market requirements in the fuel-scarce economy expected in the last decade of the twentieth century.

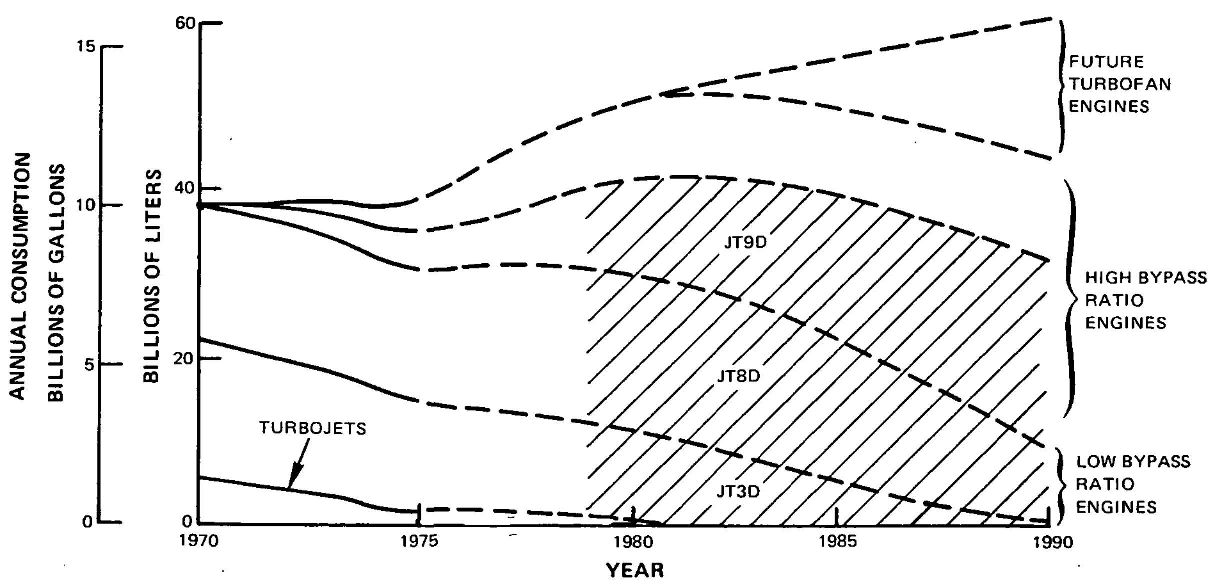



*Figure 2.0-1 Annual Fuel Consumption by U.S. Scheduled Airlines*

In addressing fuel conservation, this study program investigated the possibilities for both the current and advanced conventional turbofan engines. The study described in this report was divided into four principal tasks:

- Task I studies included the evaluation of feasible improvements that could be incorporated in the JT9D, JT8D, and JT3D engines to decrease the engine fuel consumption and direct operating costs to the airline operators.

- The Task II studies considered 1985 technology turbofan design concepts to minimize fuel consumption through the use of improved new engine components and advanced thermodynamic cycles. A wide range of thermodynamic cycles was assessed on a preliminary basis to establish the cycle requirements of the advanced technology turbofans.

- Task III studies included a more refined analysis of several promising engine cycles selected from the preceding Task II parametric analysis. Further detailed screening was conducted and an engine cycle, designated the P&WA STF 477 study turbofan, was selected for refined analysis. This engine definition embodied those technology advancements showing the greatest potential for fuel savings.

- The Task IV studies consisted of technology program formulation and recommendations directed toward current and advanced turbofan engine fuel conservation. The majority of these programs were based on the results of the Task I, II, and III evaluations. Several additional programs were also recommended as being critical to fuel conservation although they were not specifically included in these evaluations. Examples are operational performance retention programs that encompass the current and future turbofan engines and an engine controls program for the future turbofan engines.

The International System of Units is the primary system for presentation of data in this report, with the English system secondary. All calculations were made in the English system.

# 3.0 RESULTS OF STUDY

This section, which presents the significant results obtained from the study, is arranged in three parts: (1) a brief description of internal and external (installation) improvements to current commercial engines that provide fuel savings, together with an indication of the programs to accomplish these savings; (2) a brief description of concepts for reducing the fuel consumption in future (1985) turbofan engines, and (3), a brief description of performance retention programs recommended to attack the problems of short and long term performance deterioration in current and future turbofan engines.

A more detailed discussion of these results will be found in Sections 4.0 through 7.0 of this report.

## 3.1 CONCEPTS FOR REDUCING FUEL CONSUMPTION IN CURRENT TURBOFAN ENGINES

The concepts that have the potential for reducing fuel consumption in current turbofan engines are listed in Table 3.1-I. Each concept is estimated to provide a fuel savings of at least 0.4 percent when incorporated into a new production engine or retrofitted into an existing engine. The option of improving present nozzle performance or using forced mixed installations exists in new production for each of the engines. Engine exhaust emissions and noise levels are either unaffected or reduced by all of the selected modifications.

The cumulative fuel savings and direct operating cost (DOC) benefits achieved by incorporating the above concepts in current engines are shown in Table 3.1-II. For the new production case, the installation improvement with the greater fuel savings potential was used for each engine.

The individual contribution that each concept makes in terms of fuel savings and reduced direct operating costs is indicated in Figures 3.1-1 for the JT9D engine, 3.1-2 for the JT8D engine, and 3.1-3 for the JT3D engine.

### 3.1.1 Concept Summaries and Key Program Items

A brief description of each of the concepts for improving fuel consumption is given in the following paragraphs. Accompanying this description is an indication of the program recommended to develop the concept.

Experience shows that the individual modifications should be incorporated both one at a time and in groups into an engine to evaluate overall performance and stability effects. It would also be desirable to test all of the modifications simultaneously at the end of the program to evaluate the total effect on performance.

#### 3.1.1.1 Internal Engine Improvements

High Flow Capacity Fan (JT3D-7 Fan) (JT3D) - Improvements in both fan efficiency and flow capacity have been demonstrated with the JT3D-7 fan configuration. The higher flow capacity JT3D-7 fan is directly interchangeable with the present JT3D-3B configuration. No additional design or development effort is required. However, the thrust, engine pressure ratio, exhaust

TABLE 3.1-I

CONCEPTS FOR REDUCING FUEL CONSUMPTION
IN CURRENT TURBOFAN ENGINES

| | JT9D | | JT8D | | JT3D | |
|---|---|---|---|---|---|---|
| | New Prod. | Retrofit | New Prod. | Retrofit | New Prod. | Retrofit |
| **SELECTED INTERNAL ENGINE MODIFICATIONS** | | | | | | |
| • **Component Aerodynamic Improvements** | | | | | | |
| (1) High Flow Capacity Fan | --- | --- | --- | --- | X | X |
| (2) Fan Blade Performance | X | --- | X | X | X | X |
| (3) Low-Pressure Turbine Performance | --- | X | X | --- | X | --- |
| (4) Turbine Exhaust Case Strut Aerodynamic Redesign | --- | --- | X | --- | --- | --- |
| (5) Resized Turbine and Primary Exhaust Nozzles | --- | --- | --- | --- | X | --- |
| • **Material/Cooling Technique Improvements** | | | | | | |
| (6) Abradable Compressor Gaspath Seals | --- | --- | X | X | X | X |
| (7) Fan Air Cooled High-Pressure Turbine Case | X | --- | --- | --- | --- | --- |
| (8) Replacement Bearing Compartment Seals | X | --- | --- | --- | --- | --- |
| • **Structure-Mechanical Improvements** | | | | | | |
| (9) Structural Fan Exit Guide Vanes | X | --- | --- | --- | --- | --- |
| (10) Improved Compressor Interstage Cavity Sealing | --- | --- | X | --- | X | --- |
| (11) Case-Tied Low-Pressure Turbine Seals | X | --- | --- | --- | X | --- |
| **SELECTED INSTALLATION MODIFICATIONS** | | | | | | |
| • **Exhaust Nozzle Improvements** | | | | | | |
| (12) Forced Mixing of Primary and Fan Exhaust Streams | X | --- | X | --- | X | --- |
| | (or) | | (or) | | (or) | |
| (13) Replacement Exhaust Nozzles | X | X | X | X | X | X |
| **Subtotals** | | | | | | |
| **New Production** | 7 | | 7 | | 9 | |
| **Retrofit** | | 2 | | 3 | | 4 |
| **TOTALS** | | | | | | |
| **New Production** | | | 23 | | | |
| **Retrofit** | | | 9 | | | |

TABLE 3.1-II

CUMULATIVE FUEL SAVINGS AND DIRECT OPERATING COST BENEFITS

| | JT9D-7 | | JT8D-9 | | JT3D-3B | |
|---|---|---|---|---|---|---|
| | New Production | Retrofit | New Production | Retrofit | New Production | Retrofit |
| Fuel Savings from Internal Engine Improvements (%) | 2.9 | 0.5 | 2.7 | 1.3 | 4.3 | 2.3 |
| Fuel Savings from Installation Improvements (%) | 3.6 | 1.1 | 0.6 | 0.6 | 3.5 | 0.5 |
| TOTAL FUEL SAVINGS (%) | 6.5 | 1.6 | 3.3 | 1.9 | 7.8 | 2.8 |
| DOC Reductions Resulting from Internal Engine Improvements (%) | 1.4 | 0.1 | 0.8 | 0.3 | 1.9 | 0.8 |

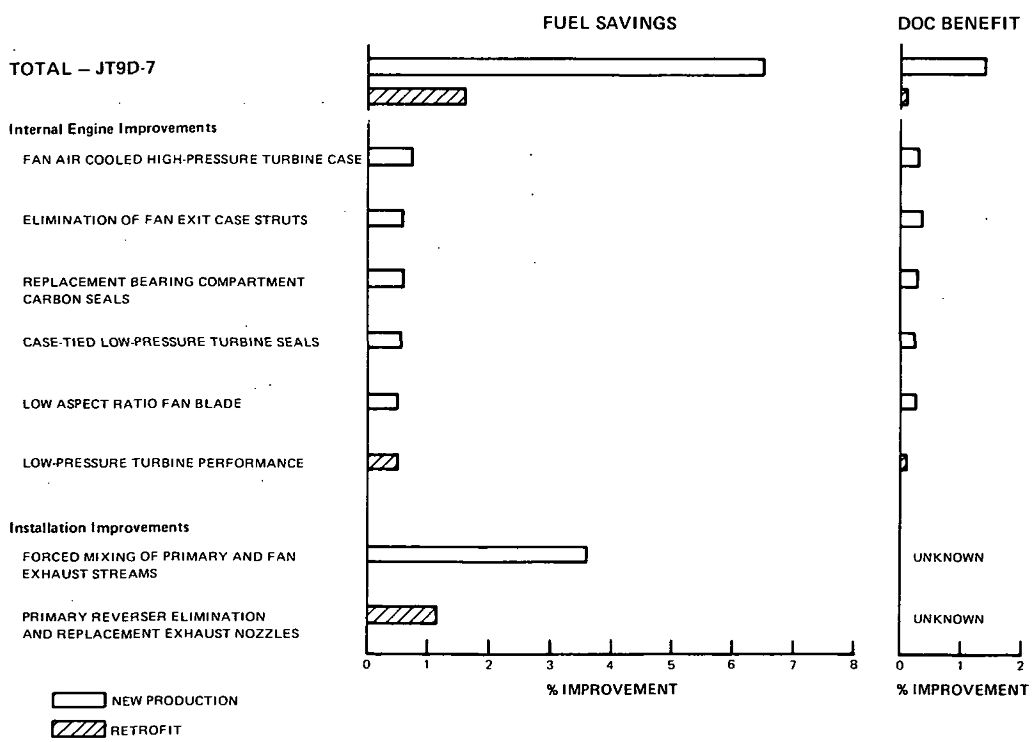



*Figure 3.1-1 Contribution of Individual Fuel Saving Concepts for the JT9D Engine*

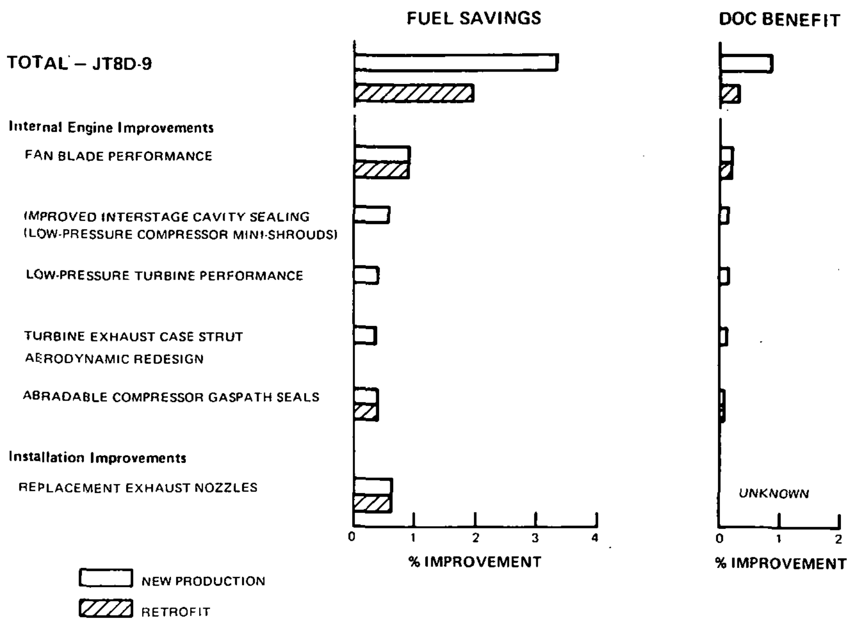



*Figure 3.1-2 Contribution of Individual Fuel Saving Concepts for the JT8D Engine*

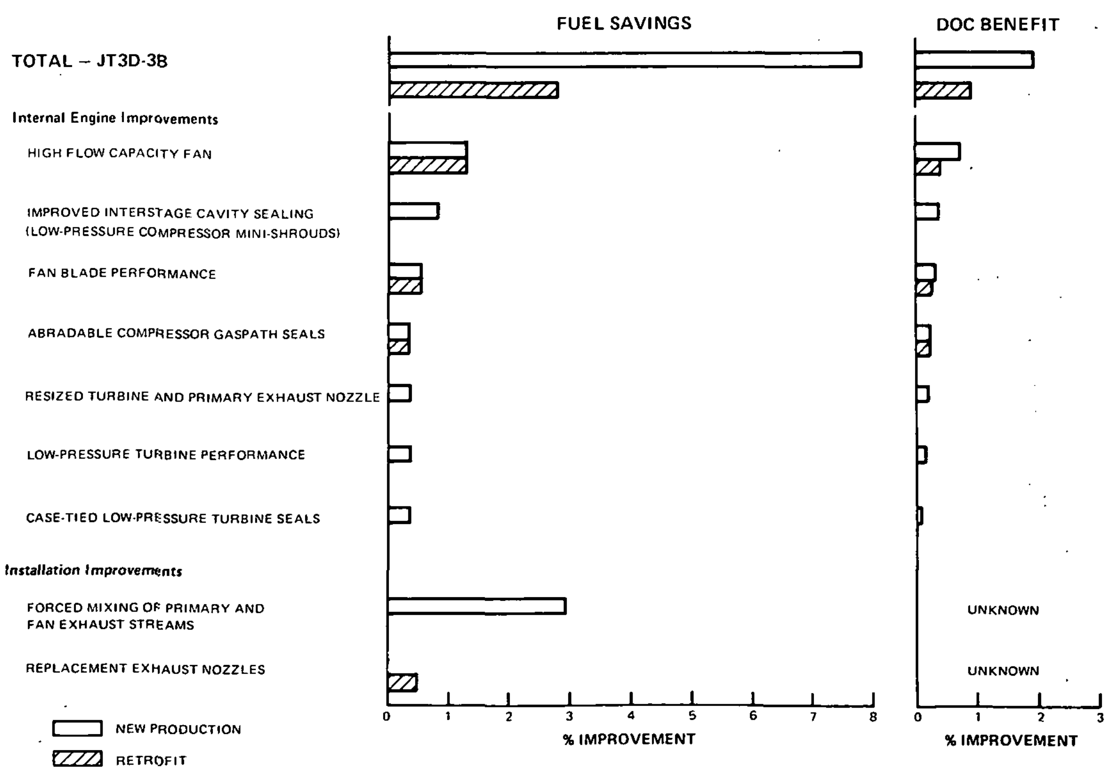



*Figure 3.1-3 Contribution of Individual Fuel Saving Concepts for the JT3D Engine*

gas temperature, and rotor speed relationships of the JT3D-3B at its rating points will be affected, requiring both recertification by the Federal Aviation Administration (FAA) and revision of engine operating manuals.

Fan Blade Performance (Chamfered-Cut Leading Edge) (JT8D, JT3D) - Appreciable gains in fan blade performance can be achieved by thinning or chamfer cutting the blade leading edge to the minimum required radius to reduce blockage effects and other aerodynamic losses. A key element in substantiating this modification is to ensure that expected performance benefits can be achieved with adequate blade structural integrity. The recommended technology verification program would involve experimental rig testing of the current configuration, followed by testing of several modifications to select the best configuration.

Fan Blade Performance (Low Aspect Ratio Fan Blade) (JT9D) - Removal of part span shrouds, made possible with the use of longer chord fan blades, leads to better fan efficiency. A 3.8 aspect ratio fan blade has been demonstrated in an experimental JT9D experimental engine. Other aspect ratios in the range of 3.8 to 4.6 (production blade) should be considered. Evaluation of the selected configuration in an experimental engine program is recommended. The scope of this effort would include an assessment of performance, stress, noise, and stability characteristics. Additional testing would focus on foreign object resistance and containment.

Low-Pressure Turbine Performance (JT8D, JT3D) - Improving the low-pressure turbine performance could contribute a substantial improvement in component efficiency. The design approach would be to optimize stage and spanwise loading distributions using controlled vortex principles and to refine the airfoil aerodynamic design. Both rig and full-scale engine test programs would be necessary to determine turbine performance effects as well as the stress characteristics of the revised airfoil design.

Sixth-Stage Turbine Blade Performance (JT9D) - Tests of various JT9D experimental engines have shown that there is a performance advantage by reducing swirl into the turbine exit guide vane. It is recommended that the design modification to the sixth-stage low-pressure turbine blade be evaluated through engine testing.

Turbine Exhaust Case Strut Aerodynamic Redesign (JT8D) - The turbine exhaust case design, evaluated under the NASA Refan Program, could be applied to the JT8D-9 engine model to recover some of the performance loss resulting from turbine exit swirl. Engine test of the redesigned case is recommended to assess the performance improvement.

Resized Turbine and Primary Exhaust Nozzles (Rematched Engine) (JT3D) - Rematching the engine for improved performance at cruise conditions could be accomplished with only minor modifications to existing hardware. The recommended modifications would be engine tested at sea level and at simulated cruise conditions for a demonstration of performance and stability effects.

Abradable Compressor Gaspath Seals (JT8D, JT3D) - Increased efficiencies in the high and low-pressure compressors can be achieved by using abradable rubstrips over the blade tips to maintain close operating clearances. This would reduce losses associated with recirculating air. Rig testing would evaluate bonding characteristics, metallography, abradability, and erosion resistance. Engine testing would be necessary to qualify the sealing technique with particular emphasis on what happens during engine transients. It would also be necessary to develop fabrication techniques to better define application approaches.

Fan Air-Cooled High Pressure Turbine Case (JT9D) - Cooling of the high pressure turbine case would allow the turbine rotors to run with tighter blade tip clearances. This would result in improved turbine efficiency. Coordination with the airframe manufacturer would be required because of the re-routing of the nacelle cooling air. Demonstration testing would follow this effort.

Replacement Bearing Compartment Carbon Seals (JT9D) - A wet-faced carbon seal system for the JT9D-7 engine Number 3 bearing compartment has been ground tested in an experimental engine. An altitude demonstration of the estimated performance effect is recommended. This type of seal gives better sealing of the bearing compartment by reducing the amount of bleed air required for the currently used labyrinth seal system.

Structural Fan Exit Guide Vanes (JT9D) - The intent of this concept is to reduce pressure losses by eliminating case struts and transferring the loads through the structural guide vanes. Analytical design studies would be necessary to predict noise, aerodynamic, and structural performance of the suggested modification. Engine testing would also be necessary to substantiate these results.

Improved Compressor Interstage Cavity Sealing (Mini-Shrouds) (JT8D, JT3D) - In conventional shrouded stator compressors, the cavities at the inner wall between the rotor and stator are large and allow secondary flow patterns to develop and interact with the primary stream. Both rig testing and experimental engine testing are necessary to substantiate a design method to reduce the cavity size. Specific attention would be directed toward demonstrating performance, surge margin, and stress levels with the selected design approach.

Case-Tied Low-Pressure Turbine Seals (JT9D, JT3D) - The concept of case-tied low-pressure turbine outer seals is a practical approach to minimizing seal clearance to reduce parasitic leakage past the rotor. A detailed thermal response study would be necessary to identify the clearance pinch point and to define the minimum allowable clearance for each seal. Design verification would be accomplished through a combination of rig testing in a simulated operating environment and a full-scale engine test.

### 3.1.1.2 Installation Improvements

Forced Mixing of Primary and Secondary Exhaust Streams (JT9D, JT8D, JT3D) - Expansion of forced mixer technology to realize the potential performance improvements would involve a combination of analytical and experimental programs. The recommended approach would be to start with a joint Pratt & Whitney Aircraft/airframe manufacturer program to assess more accurately the expected fuel savings. Various model test programs are recommended to identify the operational characteristics of forced mixing. The scope of these tests would include assessment of mixer performance for various fan and engine core flow profiles, the effect of the pylon on mixer performance, and noise. Also, an installed power model test is recommended to determine nacelle/wing/pylon interactions. Based on these results, engine testing of a selected mixer configuration is recommended.

Replacement Exhaust Nozzles (JT9D, JT8D, JT3D) – The objective of this modification is to reduce drag by recontouring the nacelle afterbody. Qualification of the fuel savings would require a combined model, full-scale, and flight test program. The model testing would involve running of isolated and installed small scale nozzles to evaluate their external drag and basic static internal performance. A full-scale engine test in an altitude facility would supplement the model test efforts and the accuracy of the internal performance data. Final verification of the actual fuel savings would be obtained through modification of the exhaust nozzles on an aircraft and flight testing of the new configuration. For the JT9D, replacing the exhaust nozzle system would include eliminating the primary reverser, thus reducing weight and aerodynamic losses.

## 3.2 CONCEPTS FOR REDUCING FUEL CONSUMPTION IN FUTURE TURBOFAN ENGINES

The concepts that have the potential for reducing fuel consumption in future engines are listed in Table 3.2-I. In comparison with current engines, where the approach is to modify or redesign existing hardware, the following listing represents technology advances beyond the JT9D-70 level that must be acquired to realize the potential benefits. The JT9D-70 technology is the most modern presently available and, in itself, represents an advancement beyond the JT9D-7 engine studied in the current engine evaluation.

A representative advanced fuel conserving turbofan engine, based on the technology advances listed in Table 3.2-I, was defined and given the Pratt & Whitney Aircraft study designation STF 477. This high bypass ratio engine is described in detail in Section 6.3.

Cycle studies showed that high pressure ratio was a major factor in lowering fuel consumption considering the limitations associated with higher pressure ratios. However, increasing pressure ratio leads to reduced-size blading in the rear stages of the compressor, increased susceptibility to case flange leakage, and higher $NO_x$ emissions through increased compressor exit temperatures. A pressure ratio of 45:1 was selected for the STF 477 as a reasonable balance. Selection of the remaining cycle parameters (1.7 fan pressure ratio, 8.0 bypass ratio, 1427°C [2600°F] maximum combustor exit temperature) was weighted toward those values favoring fuel savings and DOC benefits.

The projected fuel savings, economic benefits and noise reduction of the STF 477 engine were assessed in a three-engine medium range aircraft and a four-engine long range aircraft. Results are presented in Figure 3.2-1. The benefits shown were achieved with a 4 to 5 EPNdB reduction in total noise.

### 3.2.1 Technology Summaries and Key Program Items

Technologies for reducing fuel consumption in future engines, installed in medium or long range aircraft, and their potential benefits, are shown in Table 3.2.1-I. The benefits include the cycle improvements available with the identified technology advancements.

Following is a brief description of the recommended program for developing each technology shown in Table 3.2.1-I to a state of readiness. Achievement of significant fuel savings will also be possible with other technology advances in acoustics, full-authority electronic digital control, and maintenance cost reduction. Stringent noise requirements are likely, and methods of achieving noise levels well below the Federal Aviation Regulations Part 36 (FAR 36) will be necessary without paying a significant fuel consumption penalty. Low fuel consumption over the entire flight cycle and over the life of the engine will require advances in control and reductions in both short term and long term engine performance deterioration. The economic results of this study suggest that effort must be undertaken on specific means to make fuel conservative engines more economically attractive to purchase and maintain. Programs for these technologies are also summarized in this section. A more comprehensive description of these programs is presented in Section 7.0 of this report.

**TABLE 3.2-I**

**MAJOR TECHNOLOGY ADVANCES FORECAST FOR 1985**

| | |
|---|---|
| **Component Aerodynamic Improvements** | |
| • Fan | Elimination of Part Span Shrouds<br>Improved Airfoil Design<br>Reduced Endwall Aerodynamic Losses |
| • Compressor | Higher Stage Pressure Loading<br>Improved Blade Design<br>Reduced Tip Clearances |
| • Combustor | Improved Combustor Exit Temperature Profile<br>Reduced Emissions |
| • High-Pressure Turbine | Reduced Endwall Aerodynamic Losses<br>Reduced Tip Clearances |
| • Low-Pressure Turbine | Increased Load Factor<br>Improved Aerodynamics<br>Reduced Tip Clearances |
| **Material and Cooling Improvements** | |
| • Combustor | Advanced High-Temperature Combustor Liner Material |
| • High-Pressure Turbine | Monocrystal/Eutectic Airfoils<br>High-Temperature Protective Coatings<br>Reduced Turbine Cooling Penalty |
| **Structural-Mechanical Improvements** | |
| • Fan | Increased Tip Speed Capability |
| • Compressor | Increased Tip Speed Capability |
| • Diffuser | Improved Diffuser Design |
| • High-Pressure Turbine | Increased Speed Capability<br>Advanced Turbine Seals |

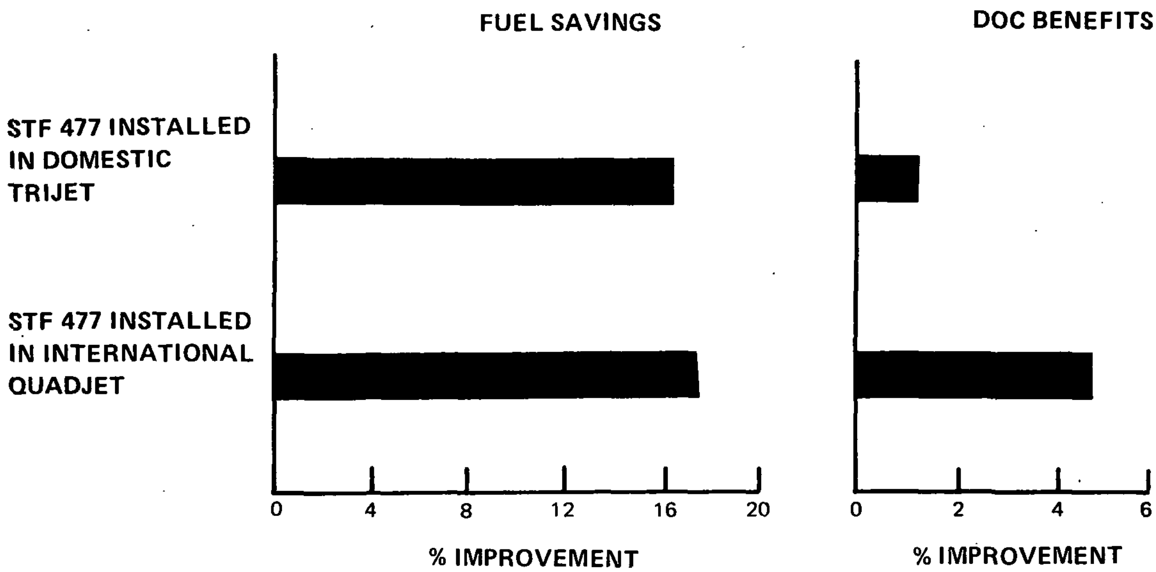



*Figure 3.2-1 Projected Installed Fuel Savings and DOC Benefits of STF 477 Engine Relative to JT9D-70 Technology*

TABLE 3.2.1-I

1985 TURBOFAN TECHNOLOGY REQUIREMENTS AND POTENTIAL BENEFITS

| | Fuel Savings Relative to JT9D-70 Technology |
|---|---|
| • Advanced High Spool | 11% |
| High Temperature Combustor and Turbine Airfoil Materials and Coatings | |
| Efficient, High Speed High Spool System | |
| • Improved Passive and Active Clearance Control Seals | 3% |
| • Advanced Low Pressure Spool | 3% |
| High Efficiency Fan | |
| High Load Factor Turbine | |
| • High Strength-to-Density Ratio Materials | 1% |
| Composites | |
| Titanium Base Alloys | |
| Nickel Alloy Disk | |

High Temperature Materials and Coatings for Combustor and Turbine Airfoils – The projected fuel savings for future turbofan engines reflect an increase in combustor liner and turbine blade metal temperatures of 83°C to 111°C (150°F to 200°F). An advanced combustor liner material and advanced turbine airfoil alloys show promise for such high temperature applications. Oxidation-erosion resistant and/or insulative coatings will also be needed for the turbine blades, vane platforms, and outer air seals. Developing this technology would require intensive metallurgical investigations and rig test efforts. Investigations would also be necessary to address the areas of fabrication and repair of high temperature materials and coatings.

Efficient High Speed High-Pressure Spool System – The combination of technological advances in the aerodynamics of the compressor, combustor, and high-pressure turbine has shown significant potential for reducing fuel consumption in future turbofan engines. Research and technology programs are required in each of these areas if the potential improvements are to be realized. An advanced high-pressure spool system would also serve as a vehicle for demonstrating new materials, advanced cooling techniques, active clearance control, and high-speed bearings and seals.

Additional analytical and test programs are recommended to reduce airfoil and endwall aerodynamic losses for maximum compressor efficiency. This effort would include testing the compressor both as an individual component and as part of a high-pressure spool system.

The recommended program for developing an advanced combustor would concentrate on reduced emissions, in conjunction with high temperature and high pressure operation. A selected low-emissions combustor concept would be evaluated in a rig and also as part of the high-pressure spool. A portion of this program would also address optimizing a diffuser aerodynamic design for integration into the high-pressure spool.

The desire for higher turbine efficiency with increased rotational speed and reduced load factor increases operating stress levels and other aerodynamic losses. Thus, the suggested turbine development program would focus on resolving these limitations, while improving performance efficiency. Component performance verification would be required by operating the turbine in a high-pressure spool engine simulating both gas path and non-gas path engine temperatures and pressure conditions.

High rotational speeds, coupled with increased pressure levels of the turbofan, require significant advances in the engine main bearings and bearing compartment seals. The technology programs recommended for these components would develop new design concepts to achieve a high speed level, while emphasizing durability.

Improved Passive and Active Clearance Control Seals – A program is recommended to develop the technology and systems to actively and passively modulate turbine and compressor blade tip clearances throughout the flight envelope. This effort would encompass assessing mechanical, pneumatic, and thermal-responsive schemes for reducing tip clearances to near zero at the cruise operating point.

Advanced Low-Pressure Spool – Technological advances in the fan/low pressure turbine rotor system could provide a fuel savings of approximately three percent relative to current turbofan engines. The principal requirements in fan technology advances are the reduction of airfoil and endwall losses without degrading aeroelastic integrity. A major part of the recommended fan program would include an evaluation of reduced losses with improved part-span shroud designs, as well as the possible elimination of part-span shrouds with the use of composite materials.

For the low-pressure turbine, the analytical verification and test of laminar flow airfoils to reduce airfoil losses are included as part of the technology program.

The use of a speed reduction gear in the low-pressure spool could offer an additional fuel savings improvement. The recommended gear technology program would address the requirements of a lightweight high efficiency component with commercial maintainability and reliability. Design and testing of a gear rig and heat rejection system would lead to a full scale test in an engine environment.

High Strength-to-Density Ratio Materials – Utilization of high strength-to-density ratio materials in future turbofan engines lies in the fuel savings resulting from the reduction in propulsive system weight. For advanced composite materials, a program concentrating on foreign object damage resistance and load transfer through joints would be formulated. This would be followed by testing of full-scale composite airfoils and a static structure.

Because high-temperature titanium alloys represent a lightweight alternative to steel and nickel base alloys, a program would be devised to test the component fabricated with this material both on an individual basis and incorporated into an engine. Similarily, the use of advanced nickel alloy high-pressure turbine disks offers an appreciable weight savings. The materials development program for this alloy would concentrate on determining the feasibility of various approaches to meet the strength requirements for advanced turbine disks.

Advanced Acoustical Technology – Achievement of significant energy savings and compliance with stringent noise requirements, such as FAR 36 minus 10 EPNdB, will require advances in fan, combustor, and turbine noise technology. An analytical program is recommended to develop an improved understanding of fan detailed aerodynamic design on discrete tone, broadband, and buzz-saw noise generation. Test programs would be required in support of the analytical study to define airfoil wake characteristics, surface fluctuations, and to evaluate analtyical models. For the combustor, the recommended program is directed toward developing analytical models of noise sources, as well as testing combustors consistent with the requirements of low emissions. Since the noise characteristics of the turbine are not well defined, a noise prediction system would be developed under the defined program. In addition, cascade tests would be recommended to define wake characteristics of high stage loading blades.

In addition to noise source reductions, improved attenuation of fan, core, and turbine source noise is required. A recommended program is directed toward increasing attenuation levels in each of these areas by at least 2 EPNdB without increasing the treated area.

Full-Authority Electronic Digital Control – A digital electronic propulsion control presents possibilities for significant fuel savings when coupled with aircraft control systems. Although Pratt & Whitney Aircraft is conducting extensive research and development in the area of digital electronic controls, an additional study program is recommended. The scope of the effort would include the definition and evaluation of the benefits of an integrated aircraft/engine control system using digital electronic engine controls and digital aircraft controls. This study program would be expanded to include demonstration testing in a suitably modified aircraft.

Reduced Maintenance Costs – The impact of designs to improve specific fuel consumption have a tendency to increase engine price and maintenance cost, thereby reducing the potential benefits of low TSFC. The area of maintenance costs requires equal effort and can produce substantial impact. A study program is recommended that would be directed toward defining the cost of turbine maintenance and conceptual design studies of low cost turbine airfoil designs, or longer life designs.

## 3.3 PERFORMANCE RETENTION PROGRAMS FOR CURRENT AND ADVANCED ENGINES

### 3.3.1 Current Engines

Load Sharing (JT9D) – Preliminary analytical studies have indicated that integrating the engine and nacelle to accomplish structural load sharing can reduce engine case and shaft deflections. These are believed to be a major cause of short term performance deterioration in modern high bypass ratio engines. Joint design studies between Pratt & Whitney Aircraft and the airframe manufacturers are recommended to define the structural configuration details of an integrated design. Structural testing of subscale prototype hardware of integrated nacelle schemes would be necessary.

Diagnostic Engine Testing (JT9D) – An engine test program to verify the cause of short term deterioration is recommended. The scope of this effort would include monitoring several operational JT9D engines to measure loads and temperatures, as well as testing engines under simulated flight loads and temperatures to investigate blade and seal clearance changes. Results would be used in the JT9D load sharing nacelle effort described in the previous paragraph.

Long Term Deterioration (JT9D, JT8D, JT3D) – Long term deterioration of JT8D and JT3D performance with accumulated service time is primarily caused by deterioration of the compressor and fan performance efficiencies. A program to address this problem would involve tests ranging from metallurgical investigations to individual component rig tests.

### 3.3.2 Advanced Engines

Operational Performance Retention – To understand the dynamics of both short and long term performance deterioration an investigatory program is first recommended to increase visibility in this area. Various rig test programs would be necessary to establish the mechanical changes responsible for long term deterioration, as well as to assess stage-to-stage variations. Since the solution to both short and long term deterioration could lead to a revised design approach to engine construction and installation, evaluation of alternative approaches to design is required to establish the favored approaches. This could involve flight testing of the resultant prototype configuration.

# 4.0 DISCUSSION OF RESULTS OF TASK I – STUDY OF CURRENT ENGINES

The primary objective of Task I was to recommend methods for reducing fuel consumption in current turbofan engines. The engines used in this study were the Pratt & Whitney Aircraft JT9D, JT8D, and JT3D, which are the major fuel consumers in commercial service. Fuel consumption and direct operating cost (DOC) were the principal criteria used in determining the overall feasibility of possible fuel conserving concepts. Fuel consumption is primarily engine-performance oriented and DOC is the investment and maintenance costs associated with incorporation of specified engine modifications, in addition to the fuel costs. Since these costs are directly incurred by the airline operators, they indicate the economic merits of each fuel savings modification.

## 4.1 CONCEPT SCREENING ANALYSIS

As the first step in this investigation, near-term technology advancements were selected and analyzed to determine their fuel savings potential. The areas surveyed included improvements to the internal engine, installation, and power management. Approximately 110 concepts having the potential for fuel savings were considered. (Several of these concepts are similar, differing only in the engine to which they are applied.) Most of these involved internal engine improvements in the areas of component aerodynamics, material/cooling techniques, and structure-mechanics. The remaining were modifications affecting the engine installation. There were no concepts identified in the category of power management which offered potential fuel savings. Forty-five concepts were applicable to the JT9D, 35 to the JT8D, and 30 to the JT3D. The following procedures were used to evaluate each concept.

- Determination of Fuel Savings – For each concept, component performance effects, such as efficiency, flow, and pressure loss changes, were estimated by using data from tests of similar modifications in other engines. With these performance estimates, changes were calculated for engine thrust specific fuel consumption (TSFC) at a given thrust level. Possible effects on engine stability, engine/nacelle weight, and aerodynamic drag were also taken into consideration. Finally, the engine weight, price, and maintenance cost changes attributed to component redesign were estimated.

  Aircraft assumptions listed in Table 4.1-I were used in further calculations to determine the fuel savings. These assumptions were also used in calculating the effect of a particular modification on DOC. The assumptions included selecting specific aircraft models representing the predominant consumers of fuel, along with various flight-service characteristics, such as flight distance and load factor. Projections of fuel and labor costs used 1974 values as a base.

  As indicated in Table 4.1-I, fuel prices were assumed for both domestic and international markets. The net fuel savings were obtained by combining the estimates of engine thrust specific fuel consumption, engine weight effects, nacelle weight and drag effects, and using the aircraft assumptions.

TABLE 4.1-I

AIRPLANE ASSUMPTIONS FOR CURRENT ENGINE EVALUATIONS

| Engine | JT9D-7 | JT8D-9 | JT3D-3B |
|---|---|---|---|
| Aircraft | 747-200 | 727-200 | 707-320 |
| Type | International | Domestic | Domestic |
| Ave. Flight Distance, km (n. mi.) | 3700 (2000) | 1300 (700) | 3700 (2000) |
| Ave. Flight Load Factor, % | 55 | 55 | 55 |
| Fuel Price, ¢/liter (¢/gal.)* | 12 (45) | 8 (30) | 8 (30) |
| Labor Rate, $/hr.* | 7.30 | 7.30 | 7.30 |

*Price estimates based on 1974 dollars.

- Determination of DOC – Airline DOC effects were established by using sensitivity factors based on the economic assumptions and other variables presented in Appendix A. The economic benefits were computed for internal engine improvements only, because the costs pertaining to nacelle and installation modifications must come from the airframe manufacturer.

The results obtained from the screening evaluations are presented in Tables 4.1-II, 4.1-III, and 4.1-IV for the JT9D, JT8D, and JT3D engine, respectively. These tables list the calculated fuel savings and DOC benefit for each fuel conserving concept. With the exception of a few items indicated in these tables, the DOC estimates were made first for new production applications, although the possibility of incorporating the modification on a retrofit basis was also considered. This approach was taken because the economics would be more favorable for implementing a modification on a new production basis.

## 4.2 REFINED ANALYSIS

The results from the screening provided the basis for selecting the more promising concepts in terms of fuel savings and DOC. The principal criteria governing this selection were a fuel savings of at least 0.4 percent and an attendant economic benefit. A 0.4 percent savings was considered the minimum practical value to demonstrate a measurable improvement when several small changes are incorporated into an engine.

A total of 23 concepts were retained for further evaluation on a new production basis. Nine remained for study on a retrofit basis because they offered a DOC benefit inclusive of fuel cost, investment and maintenance costs, and retrofit cost. In addition, concepts with a 0.4 percent or greater fuel savings potential, but requiring nacelle modifications, were retained even though the economic effects were not known.

TABLE 4.1-II

CANDIDATE FUEL CONSERVING CONCEPTS FOR THE JT9D ENGINE

| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) |
|---|---|---|
| • **Component Aerodynamic Improvements** | | |
| Low-Pressure Compressor Blade Root Sealing | 0.22 | 0.06 |
| High-Pressure Compressor Blade Root Sealing | 0.22 | 0.06 |
| High-Pressure Turbine Blade Root Sealing | 0.23 | 0.10 |
| Improved Case Flange Sealing | 0.16 | 0.06 |
| Aerodynamic Redesign of Turbine Exhaust Total Pressure Probe | 0.23 | 0.10 |
| Fourth-Stage Turbine Blade Performance | 0.06 | 0.03 |
| *Sixth-Stage Turbine Blade Performance | 0.46 | 0.22 |
| High-Pressure Spool with Reduced Rotor Windage Loss | 0.34 | 0.15 |
| Low-Pressure Spool with Reduced Rotor Windage Loss | 0 | – |
| Low Aspect Ratio Fan Blade | 0.50 | 0.26 |
| Fan Design Optimized for Cruise Conditions | 0 | – |
| Improved Fan Blade Tip Treatment (D-3A Fan) | 0.27 | 0.12 |
| Improved Fan Blade Tip Treatment (D-7 Faired-Tip Fan) | 0.28 | 0.53 |
| Improved Fan Blade Tip Clearance Control | 0.06 | −0.04 |
| Fan Blade Performance | 0.28 | 0.06 |
| Low-Pressure Compressor Blade Aerodynamic Redesign | 0.11 | 0.05 |
| Improved Aerodynamic Matching of Low-Pressure Compressor Strut and Fourth-Stage Stator Vane | 0 | – |
| Fixed Geometry Inlet Guide Vane | 0.23 | 0.10 |
| Fourth-Stage Stator Vane Aerodynamic Redesign | 0 | −0.01 |
| Revised Schedule for Variable-Geometry High-Compressor Vanes | 0.23 | 0.11 |
| Improved Aerodynamic Matching of High-Pressure Compressor Strut and Fifteenth-Stage Stator Vane | 0 | – |
| New High Pressure Ratio (10:1) High-Pressure Compressor | 2.67 | −2.44 |
| New High Pressure Ratio (11:1) High-Pressure Compressor | 3.44 | −2.15 |
| Advanced Low-Emissions Combustor Design | 0.05 | Unknown |
| Fan Exit Case Strut Aerodynamic Redesign | 0.22 | 0.11 |
| Resized Turbine and Primary Exhaust Nozzle | 0 | – |
| • **Materials/Cooling Technique Improvements** | | |
| Advanced Materials and Cooling Concepts to Reduce Turbine Cooling Air Requirements | 0.83 | −3.92 |
| Mid-Compressor Airbleed Cooled Second-Stage Turbine Stator and Seal System | 0.28 | 0 |
| Fan Air Cooled High-Pressure Turbine Case | 0.75 | 0.30 |
| Advanced Titanium Alloys in High-Pressure Compressor | 0.03 | 0.01 |
| Advanced Titanium Alloys for Sixth-Stage Low-Pressure Turbine | 0.07 | 0.02 |
| Aluminum Honeycomb Fan Discharge Case (D-20) | 0.07 | 0.07 |

*Evaluated for retrofit only.

**TABLE 4.1-II**

**CANDIDATE FUEL CONSERVING CONCEPTS FOR THE JT9D ENGINE (Continued)**

| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) |
|---|---|---|
| **• Materials/Cooling Technique Improvements (cont'd)** | | |
| Aluminum/Kevlar Fan Case (D-7) | 0.14 | 0.09 |
| Honeycomb Fan Exit Case | 0.03 | 0.03 |
| Replacement Bearing Compartment Carbon Seals | 0.59 | 0.30 |
| Number 3 Bearing Compartment Honeycomb Seals | 0.30 | −0.15 |
| **• Structural-Mechanical Improvements** | | |
| Revised First-Stage High-Pressure Turbine Outer Airseal | 0.23 | 0.10 |
| Smaller Constant Speed Drive Oil Cooler | 0.18 | Unknown |
| Reduced Fan Exit Guide Vane Density and Shorter Airfoil Chord | 0.29 | −0.08 |
| Structural Exit Guide Vanes | 0.60 | 0.35 |
| Improved Compressor Interstage Cavity Sealing | 0 | – |
| Elimination of Combustor "Z" Ring | 0.16 | 0.05 |
| Case Tied Low-Pressure Turbine Seals | 0.50 | 0.25 |
| **Installation Improvements** | | |
| **• Exhaust System Improvements** | | |
| Elimination of Primary Reverser | 0.84 | Unknown |
| Forced Mixing of Primary and Fan Exhaust Streams | 3.6 | Unknown |
| Replacement Exhaust Nozzles | 0.40 | Unknown |

TABLE 4.1-III

CANDIDATE FUEL CONSERVING CONCEPTS FOR THE JT8D ENGINE

| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) |
|---|---|---|
| **• Component Aerodynamic Improvements** | | |
| Improved Case Flange Sealing | 0.08 | −0.03 |
| Low-Pressure Turbine Performance | 0.50 | 0.21 |
| Turbine Exhaust Case Strut Aerodynamic Redesign | 0.40 | 0.13 |
| Aerodynamic Redesign of Turbine Discharge Total Pressure and Total Temperature Sensing Probes | 0.12 | 0.006 |
| High-Pressure Turbine Blade Root Sealing | 0.07 | 0.02 |
| Low-Pressure Turbine Blade Root Sealing | 0.03 | −0.02 |
| Low-Pressure Compressor Blade Root Sealing | 0.20 | 0.05 |
| High-Pressure Compressor Blade Root Sealing | 0.30 | 0.06 |
| Low-Pressure Compressor Airfoil Aerodynamic Redesign | 0 | – |
| High-Pressure Compressor Airfoil Aerodynamic Redesign | 0 | – |
| Intermediate Case Strut Matching | 0 | – |
| Fan Blade Performance | 0.94 | 0.21 |
| Reduced Part-Span Shroud Losses | 0.16 | 0.06 |
| Resized Turbine and Primary Exhaust Nozzle | 0 | – |
| Advanced Low-Emissions Combustor Design | 0.66 | Unknown |
| NASA Refan (JT8D-109) | −2.66* | −5.5* (Retrofit) |
| **• Materials/Cooling Technique Improvements** | | |
| Carbon Seal for Number 4 Bearing Compartment | 0.22 | 0.15 |
| Honeycomb Seals for Number 4 Bearing Compartment | 0 | – |
| Abradable High-Pressure Turbine Tip Seals | 0.12 | 0.09 |
| Abradable Low-Pressure Turbine Tip Seals | 0.07 | 0.35 |
| Abradable Low-Pressure Turbine Knife-Edge Seals | 0.04 | 0.10 |
| Abradable Low-Pressure Compressor Tip Seals | 0.20 | 0.05 |
| Abradable High-Pressure Compressor Tip Seals | 0.45 | 0.14 |
| Abradable Low-Pressure Compressor Knife-Edge Seals | 0.11 | 0.006 |
| Abradable High-Pressure Compressor Knife-Edge Seals | 0.09 | −0.02 |
| High-Temperature Material for Fourth-Stage Turbine Disk | 0.05 | 0.02 |
| Advanced Titanium Alloy for High-Pressure Compressor Disk and Spacer Components | 0.08 | −0.61 |
| Laminar Construction of Diffuser, Combustor, Turbine and Fan Ducts | 0.05 | 0.02 |

*Information obtained from reference 1 and evaluated for retrofit only.

TABLE 4.1-III

CANDIDATE FUEL CONSERVING CONCEPTS FOR THE JT8D ENGINE (Continued)

| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) |
|---|---|---|
| **• Structural-Mechanical Improvements** | | |
| Case-Tied Low-Pressure Turbine Seals | 0.26 | 0.09 |
| Improved Low-Pressure Compressor Interstage Cavity Sealing | 0.37 | 0.13 |
| Improved High-Pressure Compressor Interstage Cavity Sealing | 0.59 | 0.19 |
| **Installation Improvements** | | |
| **• Exhaust System Improvements** | | |
| Fan Duct Splitter Designed for Cruise Operation | 0 | – |
| Faired Fan Duct | 0.26 | 0.09 |
| Replacement Exhaust Nozzles | 0.60 | Unknown |
| Forced Mixing of Primary and Fan Exhaust Streams | 0.40 | Unknown |

TABLE 4.1-IV

CANDIDATE FUEL CONSERVING CONCEPTS FOR THE JT3D ENGINE

| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) |
|---|---|---|
| **• Component Aerodynamic Improvements** | | |
| Improved Case Flange Sealing | 0.13 | 0.02 |
| Low-Pressure Turbine Performance | 0.40 | 0.19 |
| Turbine Exhaust Case Strut Aerodynamic Redesign | 0 | – |
| Aerodynamic Redesign of Turbine Discharge Total Pressure and Total Temperature Sensing Probes | 0.09 | 0.02 |
| High-Pressure Turbine Blade Root Sealing | 0.03 | 0.009 |
| Low-Pressure Turbine Blade Root Sealing | 0.03 | −0.04 |
| Low-Pressure Compressor Blade Root Sealing | 0.43 | 0.18 |
| High-Pressure Compressor Blade Root Sealing | 0.13 | 0.02 |
| Low-Pressure Compressor Airfoil Aerodynamic Redesign | 0.21 | 0.10 |
| High-Pressure Compressor Airfoil Aerodynamic Redesign | 0.07 | 0.03 |
| Fan Blade Performance | 0.67 | 0.27 |
| Reduced Part-Span Shroud Losses | 0.13 | 0.07 |
| Improved Aerodynamic Matching of Fan Exit Guide Vane | 0.07 | 0.03 |
| High Flow Capacity Fan | 1.52 | 0.75 |
| Advanced Low-Emissions Combustor Design | 0.45 | Unknown |
| NASA Refan (JT3D-9) | 0.66* | −10.4 (Retrofit) |
| Resized Turbine and Primary Exhaust Nozzle | 0.45 | 0.22 |
| **• Materials/Cooling Technique Improvements** | | |
| Abradable High-Pressure Turbine Tip Seals | 0.03 | 0.06 |
| Abradable Low-Pressure Turbine Tip Seals | 0.08 | 0.08 |
| Abradable Low-Pressure Turbine Knife-Edge Seals | 0.06 | 0.09 |
| Abradable Low-Pressure Compressor Tip Seals | 0.42 | 0.18 |
| Abradable High-Pressure Compressor Tip Seals | 0.20 | 0.09 |
| Abradable Low-Pressure Compressor Knife-Edge Seals | 0.12 | 0.02 |
| Abradable High-Pressure Compressor Knife-Edge Seals | 0.04 | −0.02 |
| Advanced Disk Materials | 0 | – |
| **• Structural-Mechanical Improvements** | | |
| Case-Tied Low-Pressure Turbine Seals | 0.40 | 0.09 |
| Improved Low-Pressure Compressor Interstage Sealing | 0.84 | 0.37 |
| Improved High-Pressure Compressor Interstage Sealing | 0.20 | 0.04 |
| **Installation Improvements** | | |
| **• Exhaust Nozzle Improvements** | | |
| Replacement Exhaust Nozzles | 0.54 | Unknown |
| Mixed Flow of Primary and Fan Exhaust Streams | 3.00 | Unknown |

*Information obtained from reference 1 and evaluated for retrofit only.

Several concepts, such as the new high pressure ratio compressor and the NASA refan, were eliminated from further consideration although they offered substantial fuel savings potential. These savings were counteracted by a high DOC penalty associated with extensive component modification.

The retrofit and new production benefits associated with the concepts that were retained for further study are shown in Tables 4.2-I, 4.2-II, and 4.2-III for the JT9D, JT8D, and JT3D engines, respectively. P&WA experience shows that when numerous changes are incorporated in an engine concurrently, they often produce less than the algebraic sum of the individual effects. This is because of such factors as adverse interactions among the individual changes. Therefore, the probable fuel savings were calculated for each concept using statistical methods. Generally, the probable savings were slightly lower than those initially obtained in the screening evaluations. These statistically determined values were used to ascertain the cumulative savings achieved by integrating these concepts into a production engine because they were considered more indicative of actual improvement. The standard deviations for each concept (not listed) were combined with an estimate of expected test accuracy during a combined demonstration of the concepts to arrive at the total fuel savings and the uncertainty shown for each grouping in the tables.

The cumulative new production fuel savings (nominal values) provided by concepts selected for the JT9D-powered aircraft is 6.5 percent. Approximately half of this total (2.9 percent, Table 4.2-I) is directly attributed to internal engine modifications. In this category, the major savings are attained with the use of a fan-air-cooled high-pressure turbine case. The remainder of the 6.5 percent savings (3.6 percent) is a result of forced mixing of primary and fan exhaust streams. The mixing benefit was calculated at Mach 0.85, 10.7 km (35,000 ft) altitude at 85 percent Maximum Cruise Rating as a representative engine operating cruise condition. The calculations are based on the thermodynamic conditions of the presently separate exhaust stream engine. Ongoing evaluations of the mixing potential of the JT9D are considering both a range of power settings and the engine rematch which occurs with a common nozzle exhaust system. When TSFC, weight, and nacelle drag differences were taken into account, the most recent studies indicate a mixer fuel savings potential which varies with power setting at the reference flight conditions as follows: 4.3 percent at 100 percent Maximum Cruise Rating decreasing to 2.8 percent at 85 percent Maximum Cruise Rating and to 2.5 percent at 80 percent Maximum Cruise Rating. The reduced benefit at the 85 percent power point relative to the baseline study is a result of a fan rematch of the mixed exhaust engine required to provide adequate high power stability margin. The translation of these cruise savings into overall energy efficiency would entail determination of the aerodynamic and structural engine-to-airframe integration effects and an integrated mission fuel consumption analysis which considers the take-off, climb, and descent segments in addition to the cruise leg. The accrued fuel savings for the JT8D and the JT3D powered aircraft (Tables 4.2-II and 4.2-III) are 3.3 and 7.3 percent, respectively, in each case using the installation improvement with the greatest new production fuel savings.

TABLE 4.2-I

CUMULATIVE SAVINGS FOR JT9D-7 POWERED B747-200 AIRCRAFT

| | Boeing 747 Benefits | | | |
|---|---|---|---|---|
| | Retrofit | | New Production | |
| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) | Fuel Savings (%) | DOC Benefit (%) |
| *Fan Air Cooled High-Pressure Turbine Case | – | – | 0.7 | 0.30 |
| Structural Fan Exit Guide Vanes | – | – | 0.6 | 0.34 |
| *Replacement Bearing Carbon Seals | – | – | 0.6 | 0.21 |
| *Case Tied Low-Pressure Turbine Seals | – | – | 0.5 | 0.25 |
| Low Aspect Ratio Fan Blade | – | – | 0.5 | 0.30 |
| *Sixth-Stage Turbine Blade Performance | 0.5 | 0.14 | – | – |
| Totals | 0.5±0.5 | 0.14 | 2.9±1.1 | 1.40 |
| **Installation Improvements** | | | | |
| Forced Mixing of Primary and Fan Streams | – | | 3.6±0.8 | |
| or | | | | |
| Replacement Exhaust Nozzles and Primary Reverser Elimination | 1.1±0.6 | | 1.1±0.6 | |

*Items already included in JT9D-70 engine model design.

TABLE 4.2-II

CUMULATIVE SAVINGS FOR JT8D-9 POWERED B727-200 AIRCRAFT

| | Boeing 727 Benefits | | | |
|---|---|---|---|---|
| | Retrofit | | New Production | |
| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) | Fuel Savings (%) | DOC Benefit (%) |
| Fan Blade Performance (Chamfered Leading Edge) | 0.9 | 0.21 | 0.9 | 0.24 |
| Improved Compressor Interstage Cavity Sealing (Mini-Shrouds) | – | – | 0.6 | 0.19 |
| *Low-Pressure Turbine Performance | – | – | 0.4 | 0.18 |
| *Turbine Exhaust Case Strut Aerodynamic Redesign | – | – | 0.4 | 0.12 |
| *Abradable Compressor Gaspath Seals | 0.4 | 0.10 | 0.4 | 0.11 |
| Totals | 1.3±1.0 | 0.31 | 2.7±1.4 | 0.84 |
| **Installation Improvements** | | | | |
| *Forced Mixing of Primary and Fan Exhaust Streams | – | | 0.4±0.6 | |
| or | | | | |
| Replacement Exhaust Nozzle | 0.6±0.8 | | 0.6±0.8 | |

*Items already included in JT8D-200 series engine model design.

TABLE 4.2-III

CUMULATIVE SAVINGS FOR JT3D-3B POWERED B707-320 AIRCRAFT

| | Boeing 707 Benefits | | | |
|---|---|---|---|---|
| | Retrofit | | New Production | |
| Internal Engine Improvements | Fuel Savings (%) | DOC Benefit (%) | Fuel Savings (%) | DOC Benefit (%) |
| High Flow Capacity Fan | 1.3 | 0.37 | 1.3 | 0.66 |
| Improved Compressor Interstage Cavity Sealing (Mini-Shrouds) | – | – | 0.8 | 0.36 |
| Fan Blade Performance (Chamfered Leading Edge) | 0.6 | 0.23 | 0.6 | 0.26 |
| Abradable Compressor Gaspath Seals | 0.4 | 0.15 | 0.4 | 0.16 |
| Resized Turbine and Primary Exhaust Nozzles | – | – | 0.4 | 0.19 |
| Low Pressure Turbine Performance | – | – | 0.4 | 0.21 |
| Case Tied Low-Pressure Turbine Seals | – | – | 0.4 | 0.09 |
| Totals | 2.3±1.1 | 0.75 | 4.3±1.6 | 1.93 |
| **Installation Improvements** | | | | |
| Forced Mixing of Primary and Fan Exhaust Streams | – | | 3.0±0.8 | |
| or | | | | |
| Replacement Exhaust Nozzles | 0.5±0.5 | | 0.5±0.5 | |

These fuel savings could be achieved without increasing either engine noise or exhaust emission levels. Forced mixing of primary and fan streams could provide a measurable reduction in engine jet noise. The aircraft used in these evaluations are representative of the majority of P&WA-powered commercial airplanes. It is most likely that other models would demonstrate similar environmental trends with these fuel-conserving concepts.

The DOC benefits were shown to be quite small for introducing internally modified engines into fleet service. For example, the DOC for a new production JT9D engine is reduced by 1.4 percent when considering the effects of fuel cost, engine price, and maintenance costs. As indicated in Table 4.2-I, this benefit is only 0.14 percent on a retrofit basis. As a result of the relatively small economic benefits, there may not be sufficient incentive for engine manufacturers to pursue the development of fuel conserving technology in view of the substantial development costs. Consequently, government-sponsored research and technology programs may be a necessary first step to stimulate development of this technology.

## 4.3 DESCRIPTION OF SELECTED FUEL CONSERVING CONCEPTS

In this section, each of the selected concepts is described in detail. The concepts are arranged according to internal engine improvements and installation improvements. The internal engine improvements are further delineated into the categories of component aerodynamic improvements, materials/cooling improvements, and structural-mechanical improvements.

### 4.3.1 Internal Engine Improvements – Aerodynamic

#### 4.3.1.1 High Flow Capacity Fan (JT3D)

The JT3D-7, with its higher flow capacity and higher efficiency fan at constant thrust, could be used to improve the earlier JT3D-3B model performance.

#### 4.3.1.2 Fan Blade Performance (Chamfered Leading Edge) (JT8D, JT3D)

Experimental testing has demonstrated that chamfering the leading edge of fan blades improves the performance of the fan. This performance improvement is attributed to two factors. One is that sharpening or thinning the leading edge reduces the basic aerodynamic losses in the supersonic Mach number region. The other is that machining produces an overcamber of the leading edge for better efficiency in the part power or subsonic operating region.

The modification is accomplished by removing material from the suction surface of the blade to form an approximate 0.17-radian (10°) chamfer at the leading edge. Following this operation, the leading edge is reworked back to the minimum leading edge radius required and the cut is contoured to remove any surface irregularities. To ensure adequate structural integrity, the leading edge is not machined below the basic minimum radius. Performance estimates for the machined blades were derived from testing of JT8D and JT3D fan blades.

4.3.1.3 Low-Pressure Turbine Performance (JT8D, JT3D)

The JT8D and JT3D low-pressure turbines were designed and developed during the early 1960's. The stage loading distributions were optimized using a design system based on free vortex principles. During the past years, these engines have undergone numerous changes to improve performance. The turbines, however, have not been significantly modified from the original design.

Turbine technology has made considerable advancements since the engines were designed. The application of non-free vortex flow patterns, in conjunction with the ability to account for the effects of streamtube curvature and spanwise entropy gradients, offers an improved method for optimizing both stage and spanwise loading distributions. In addition, advances in computer technology have made it possible to design improved airfoil contours by evaluating the aerothermodynamic behavior of surface boundary layers. Current technology provides the turbine designer with an accurate evaluation of the airfoil surface pressure distribution, especially along the "uncovered" portions of the airfoil that exist in the region of the leading and trailing edges, where adverse pressure gradients occur. The finite-difference airfoil boundary layer program utilizes the pressure distribution to predict the behavior of laminar, transitional, and fully turbulent boundary layers. This allows the turbine designer to avoid aerodynamic flow separation and provide an optimized airfoil contour.

The aforementioned design tools were not available during the design of either the JT8D or JT3D low-pressure turbines. Turbine performance could therefore be improved by optimizing stage and spanwise loading distributions using controlled vortex principles and incorporating improved airfoil contours.

4.3.1.4 Turbine Exhaust Case Strut Aerodynamic Redesign (JT8D)

During the recent JT8D Quiet Engine Program, it was noted that the turbine exit guide vane retained approximately 0.14 radian (8°) of residual swirl in the inner region of the exit flow stream. The current exit guide vane was originally designed for the JT8D-1 engine. It has a constant section airfoil and, because there are only four of these vanes, each essentially behaves as an isolated airfoil. The increased swirl into the guide vane root region in the JT8D-9 engine promotes an aerodynamic stall condition in this region, thus leaving residual swirl in the flow stream. This results in an approximate 0.5 percent loss in total engine thrust. Redesigning the airfoil with an overcambered leading edge and reduced gap/chord by using eight instead of four airfoils can remove the swirl and improve engine performance.

4.3.1.5 Low Aspect Ratio Fan Blade (JT9D)

The two part-span shrouds on the present JT9D fan blade cause an efficiency loss resulting from the drag of the shroud and interference drag between the shroud and blade. By reducing the blade aspect ratio from the present 4.6 level to 3.8, it is structurally possible to remove one of the two part-span shrouds for a favorable efficiency gain. Although a fan weight increase of 60.8 kg (134 lbm) is incurred, the improvement in fan efficiency could have a much greater effect on fuel savings potential.

#### 4.3.1.6 Sixth-Stage Turbine Blade Performance Enhancement (JT9D)

Results of various JT9D experimental engine tests show that performance can be improved by reducing swirl into the turbine exit guide vane. By revising the leading edge camber distribution of the preceding sixth-stage turbine blade, a more optimum blade exit distribution can be obtained to reduce exit guide vane pressure losses without adversely affecting low-pressure turbine performance.

#### 4.3.1.7 Resized Turbine and Primary Nozzle (Rematched JT3D Engine)

Cruise TSFC can be reduced by rematching the engine so that the fan and compressor operate at lower rotor speeds (reduced airflow) where the efficiencies are higher. The reduced airflow tends to require a higher turbine inlet temperature to maintain thrust, but the higher fan and compressor efficiencies more than offset this effect. The turbine nozzle and primary exhaust nozzle areas would be changed, but the fan nozzle area would remain unchanged.

### 4.3.2 Internal Engine Improvements – Materials/Cooling Technique

#### 4.3.2.1 Abradable Gaspath Compressor Seals (JT8D, JT3D)

Close blade tip clearances are desired for increased compressor efficiency, but the possibility of tip-to-case rubbing is a potential problem area. If the outer case is not concentric, rubbing will wear the blade tips and increase the average tip clearance. The use of abradable (sacrificial) rubstrips would allow the blades to rub into the outer wall without blade wear. As a result, closer clearances can be maintained. The most practical approach for incorporating abradable rubstrips in service engines is to either spray or bond the abradable material to the existing outer wall surface over the blade tip. Performance estimates for abradable tip seals are based on decreasing the average tip clearance to one percent of span with a sprayed nickel-graphite material.

#### 4.3.2.2 Fan Air Cooled High-Pressure Turbine Case (JT9D)

At cruise conditions, the tip gaps of the JT9D high-pressure turbine blades are larger than desirable. Cold clearances cannot be reduced, however, because of a transient pinch point (see Section 4.3.3.2). To take advantage of the improved fuel consumption associated with decreased tip gap, the turbine case can be externally cooled with fan air, causing the case and airseals to contract slightly, reducing the blade tip gap. To accomplish this, fan air would be passed through holes to impinge on the outer case wall. Operation of this cooling could be completely automatic through an electrically actuated valve that reacts to cruise altitude and high-pressure spool rotor speed. A slight nacelle modification would be required to channel the cooling air. The turbine compartment ventilation system could be affected because the fan air would be exhausted into this region after cooling the turbine case. The case would be cooled only during cruise operation (accounting for most flight time) because a complex control system would be required for operation of this concept at other flight conditions.

#### 4.3.2.3 Replacement Bearing Compartment Carbon Seals (JT9D)

The Number 3 bearing compartment in the JT9D-7 engine has a buffer seal arrangement consisting of three adjacent labyrinth seals. The innermost seal creates a pressure drop between the seventh-stage compressor air and the bearing compartment. The inner cavity is formed by another labyrinth seal with a pressure drop such that its leakage must be discharged into the exhaust duct. Finally, a third outer labyrinth seal is used to buffer this inner cavity from the compressor discharge air. This compressor discharge air mixes with the seventh-stage compressor air and is expelled into the exhaust duct, resulting in a performance loss. A replacement carbon seal eliminates the need for the two inner labyrinth seals, in addition to the seventh-stage compressor air requirement, permitting the compressor exit buffer air to be used for useful work output in the low-pressure turbine.

### 4.3.3 Internal Engine Improvements – Structural-Mechanical

#### 4.3.3.1 Improved Interstage Cavity Sealing (Mini-Shrouded Compressor Stators) (JT8D, JT3D)

In conventional shrouded stator compressors, the cavity sizes at the inner wall between the rotor and stator have a significant effect on compressor performance. Large cavities allow secondary flow patterns to develop. These interact with the primary flow stream and reduce compressor performance. The effect of recirculation can be diminished by reducing the cavity volume or reducing the cavity/flowpath opening. In the evaluation of this concept it was assumed that the inner seals were relocated closer to the flowpath to reduce the cavity size. This required redesigning the rotor disks, spacers, stator inner shrouds, and diaphragms. Alternative approaches that may be considered to lessen the extent of redesign include reducing the axial gaps between blade platforms and stator inner shrouds and installing a lightweight filler into these cavities.

#### 4.3.3.2 Case-Tied Low-Pressure Turbine Seals (JT9D, JT3D)

The term "case-tied" is used to describe an arrangement in which seals are mechanically retained to the turbine outer case structure. With this approach, the seal is forced to respond thermally in concert with the case to achieve a minimum clearance. In establishing the new clearances, the pinch point must be identified. The pinch point is defined as the occurrence of minimum clearance between the seal knife edge and seal bond, and involves the effects of thermal gradients, tolerances, and deflections due to normal loadings that could be expected on a once-per-flight basis. While often found during a transient condition (acceleration-deceleration), the pinch point can also occur at a stabilized power setting such as cruise.

#### 4.3.3.3 Structural Fan Exit Guide Vanes (JT9D)

The JT9D engine contains a row of fan exit guide vanes to remove the swirl from the flow stream, and a row of fan case struts to carry structural loads. Preliminary studies indicate that it is possible to combine these two functions in one row of airfoils and thereby eliminate the strut and the attendant aerodynamic losses. The revised configuration uses a lower aspect ratio fan blade than currently in production to maintain the proper relationship between the number of blades and vanes for acoustic considerations.

### 4.3.4 Installation Improvements – Exhaust Nozzle

4.3.4.1 Forced Mixing of Primary and Fan Exhaust Streams (JT9D, JT8D, JT3D)

Installation of a mixing device in the unmixed primary and fan streams would distribute the propulsive energy more uniformly, resulting in reduced cruise TSFC. There might, however, be an increase in nacelle drag and weight relative to the base engine which could partially offset the potential benefits.

4.3.4.2 Replacement Exhaust Nozzles (JT9D, JT8D, JT3D)

A study of the exhaust nozzle has determined that replacement of the present convergent fan nozzle with a convergent-divergent nozzle would reduce fuel consumption. In the JT9D and JT3D engines, which have a nonmixed flow exhaust configuration, reduction in nacelle drag is expected from the use of the convergent-divergent nozzle, resulting in reduced fuel consumption from both the lower thrust required and the lower TSFC at the lower cruise level. In the mixed flow JT8D engine with its relatively high exhaust expansion ratio at cruise conditions, performance improvement would result from the low area ratio of the convergent-divergent nozzle. Take-off thrust would be slightly compromised, however, because of a loss in nozzle performance at low expansion ratios at take-off conditions.

4.3.4.3 Elimination of Primary Reverser (JT9D)

Improved JT9D performance can be obtained by removing the primary reverser and redesigning the aerodynamic contour of the primary nozzle section. The contour revisions consist of shortening the primary plug, incorporating maximum angles for the engine afterbody to provide the shortest and lightest configuration, and modifying the afterbody to reduce the Mach number and pressure loss in the primary duct. This concept would be applied in conjunction with the replacement exhaust nozzle above.

## 5.0 DISCUSSION OF RESULTS OF TASK II – STUDY OF FUTURE ENGINES

The objective of Task II was to derive conceptual designs for future turbofan engines which provide significant reductions in fuel consumption. A broad range of thermodynamic cycles was studied. This study included preliminary estimates of engine performance, weight, price, maintenance costs, and resulting effect on airplane performance and operating economics. The advanced technology turbofan cycles which are defined embody technological advances forecast to be achievable for engine developments initiated in 1985 and operational in the 1990's.

### 5.1 TECHNOLOGY FORECAST FOR 1985

Recent studies and analyses have made it possible to forecast technological improvements which may be achieved by 1985 and to relate these improvements to reductions in engine fuel consumption. Improvements are expected in component aerodynamics, material/cooling technology, and structure-mechanics.

#### 5.1.1 Component Aerodynamic Improvements

As shown in Figure 5.1.1-1, the potential exists for significant TSFC improvements with increasing component aerodynamic efficiencies. Increases in adiabatic efficiency of one to three percent are forecast for each of the four major engine rotating components by 1985. If a one percent increase in efficiency is realized for each of these components, a three percent improvement in cruise TSFC would result. If a three percent improvement in efficiency is realized for each of the four components, the TSFC would be improved by over eight percent. Following are some of the specific component aerodynamic improvements which are forecast for 1985.

5.1.1.1 Fan

Projected advances in fan technology will permit reduction of airfoil and endwall losses without degrading aeroelastic integrity. More efficient controlled-shock blading will be used in place of the present multiple circular arc blading to reduce shock losses. Improvement or elimination of interblade shrouds used to control blade flutter will reduce losses in these regions of the gas path.

5.1.1.2 Compressor

Higher compressor rotor speeds and stage loadings will reduce the number of stages and attendant stage losses. Tighter gas path sealing at blade tips and other leakage paths will further reduce losses. It should be possible to control blade tip clearances in the compressor by active clearance control. This involves moving the blade-tip seals by thermal, pneumatic, or mechanical means to vary the clearance between rotating and stationary engine parts to suit each engine operating condition.

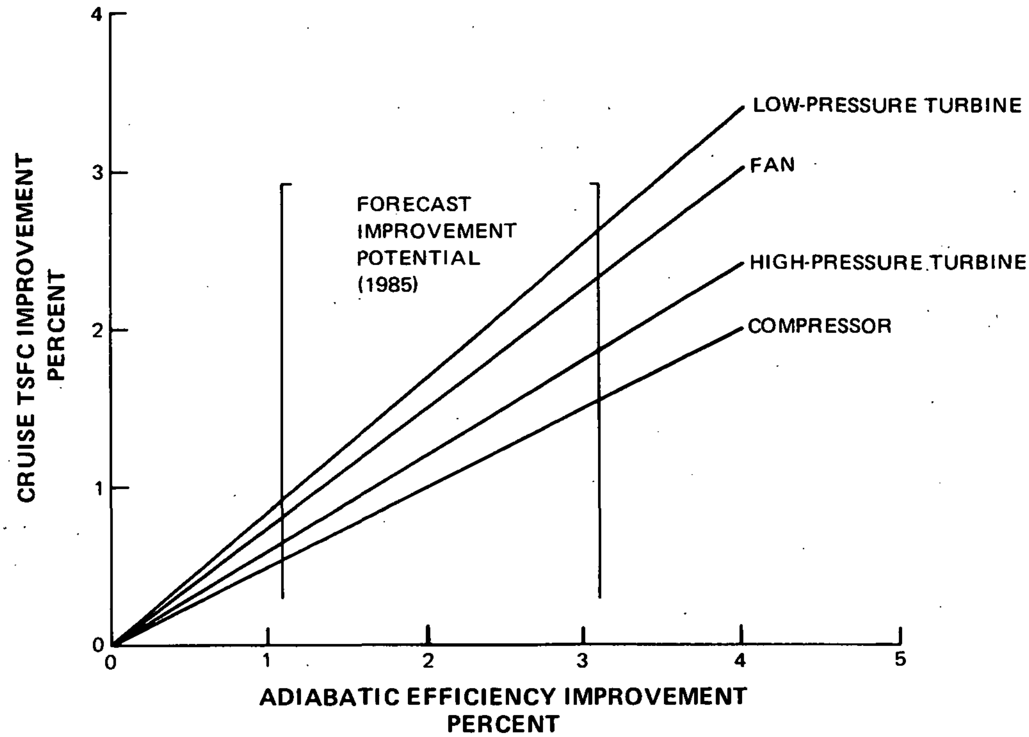



*Figure 5.1.1-1 TSFC Potential With Projected Component Aerodynamic Improvements*

5.1.1.3 Combustor

A high-temperature, high-pressure low-emissions combustor with a low-loss inlet diffuser will be matched to the compressor and turbine.

5.1.1.4 High-Pressure Turbine

The high-pressure turbine will have a higher efficiency with an increased rotor speed. New, lighter-weight blade designs will reduce the centrifugal load on the blades caused by the higher rotor speeds. Losses will be reduced by tighter gas path sealing and reduced disk windage (gas-disk friction) effects. Nondeteriorating, static abradable seals of metallic and ceramic materials, as well as improved rotating component abrasive materials suitable for use with the abradable seals, are forecast. As in the compressor, active clearance control may be used in the high-pressure turbine to control blade tip clearances.

5.1.1.5 Low-Pressure Turbine

Projected technology advances will provide a low-pressure turbine with a high load factor to minimize the number of stages and reduce stage losses. Laminar flow airfoils will reduce the blading losses. Active clearance control will be utilized to minimize running clearances.

### 5.1.2 Material/Cooling Technology Improvements

Studies indicate that a potential increase of 83°C to 111°C (150°F to 200°F) in metal temperature capability in the engine hot section can be achieved by 1985. This will result in a cruise TSFC improvement of six to eight percent (Figure 5.1.2-1). This improvement can be obtained without reducing the life of parts. Some of the technology improvements in metal temperature capability forecast for 1985 are discussed below.

#### 5.1.2.1 Advanced High Temperature Burner Liner Materials

Advanced sheet alloys have the potential for higher creep strength and higher metal temperature operating levels than the best burner liner materials now in use. Available data permit projections in metal temperature capability beyond a 111°C (200°F) increase relative to present day burner liner materials, such as Hastelloy X.

#### 5.1.2.2 Monocrystal/Eutectic Turbine Airfoil Alloys

Monocrystal/eutectic alloys and improved airfoil coatings show promise for providing higher strength, higher temperature capability in turbine blades. One nickel-columbium eutectic alloy evaluated, when combined with an improved high-temperature coating, has the potential for either a 50 percent increase in blade design stress or a 56°C to 111°C (100°F to 200°F) increase in metal temperature.

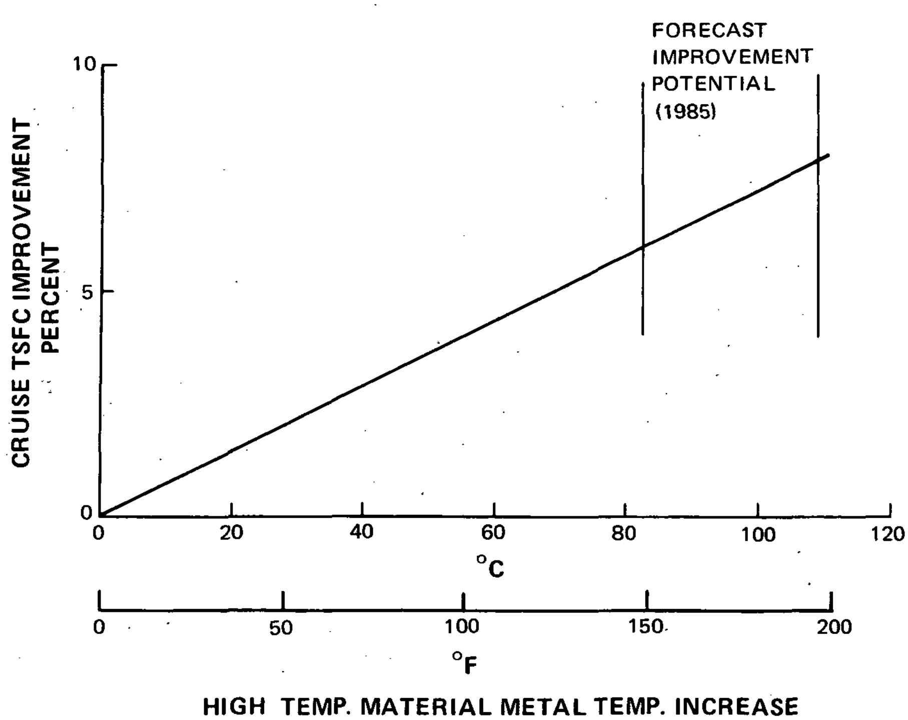



*Figure 5.1.2-1 TSFC Potential With Projected Materials/Cooling Technology Improvements*

#### 5.1.2.3 Ceramic Thermal Barrier Coatings

Ceramic materials have a temperature capability at least 278°C (500°F) greater than their metal-alloy counterparts. The low conductivity of ceramics provides an insulating effect on metal material used as a backing under the ceramic. Possible applications include vane platforms, turbine air seals, and turbine blades.

### 5.1.3 Structural-Mechanical Improvements

Rotor speeds well beyond current levels will be required to take advantage of projected improvements in aerodynamics and materials. These increased speeds will, in turn, require improvements in several structural-mechanical areas, two examples of which are lightweight turbine blades and advanced bearings and seals.

#### 5.1.3.1 Lightweight Turbine Blades

High rotor speeds in the turbine will require lightweight blades to reduce centrifugal stresses on the blade attachments. The use of advanced titanium blade alloys in place of the nickel-based alloys in the aft stages is one possibility.

#### 5.1.3.2 Advanced Bearings and Seals

Higher rotor speeds coupled with increased pressure levels will require significant advances in main engine bearings and bearing compartment seals. Bearing DN levels (bearing bore diameter times speed – mm X rpm) approaching 3,000,000 and main seal face speeds of 180 m/sec (600 ft/sec) are examples.

## 5.2 CYCLE SCREENING ANALYSIS

Using the 1985 technology forecast as a base, 26 advanced engine cycles were selected (Table 5.2-I) to evaluate fuel savings and economic benefits. The cycle parameters included overall pressure ratios ranging from 20:1 to 60:1, combustor exit temperatures from 1204°C to 1760°C (2200°F to 3200°F), fan pressure ratios from 1.3 to 1.8, and bypass ratios from 5 to 20. For each combination of cycle (overall) pressure ratio, combustor exit temperature, and fan pressure ratio selected as independent variables, bypass ratio was calculated to minimize engine fuel consumption. For the nonmixed exhaust engines, this was accomplished by setting the cruise design point duct-to-engine stream exhaust jet velocity ratio at a 0.74 level as determined by analysis. For the mixed exhaust engine studied, a lower ratio was required to maximize the mixing benefit. The resulting relationships among the various cycle variables are shown schematically in Figure 5.2-1. Maximum combustor exit temperature and fan pressure ratio variations exert the most powerful influence on bypass ratio with only a secondary effect from cycle pressure ratio; what effect there is reflects generally lower available transfer energy in the low-pressure turbine as cycle pressure ratio is increased.

# TABLE 5.2-I

## ADVANCED TURBOFAN ENGINE CYCLES SELECTED FOR STUDY

**Arranged in order of increasing fan pressure ratio and cycle pressure ratio**

| Cycle No. | Fan Pressure Ratio | Cycle Pressure Ratio | Bypass Ratio | Maximum Combustor Exit Temperature (°C) | (°F) | TSFC Maximum Cruise Rated* (kg/hr/N) | (lbm/hr/lbf) | Uninstalled Thrust, Sea Level Static Standard Day Takeoff (N) | (lbf) | Engine Weight (kg) | (lbm) | Fan Case Diameter (m) | (in.) |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 24 | 1.3 | 30 | 19.1 | 1538 | 2800 | 0.0599 | 0.587 | 129800 | 29181 | 2350 | 5180 | 2.53 | 99.7 |
| 25 | 1.3 | 40 | 20.5 | 1538 | 2800 | 0.0578 | 0.567 | 130700 | 29382 | 2540 | 5590 | 2.61 | 102.8 |
| 26 | 1.3 | 50 | 18.9 | 1538 | 2800 | 0.0571 | 0.560 | 130060 | 29238 | 2730 | 6010 | 2.58 | 101.4 |
| 14 | 1.4 | 30 | 11.9 | 1371 | 2500 | 0.0606 | 0.594 | 122960 | 27642 | 2260 | 4990 | 2.24 | 88.0 |
| 15 | 1.4 | 30 | 15.5 | 1538 | 2800 | 0.0602 | 0.590 | 125050 | 28112 | 2070 | 4570 | 2.29 | 90.3 |
| 16 | 1.4 | 30 | 20.1 | 1760 | 3200 | 0.0605 | 0.593 | 127970 | 28769 | 2110 | 4650 | 2.37 | 93.2 |
| 6 | 1.4 | 35 | 15.2 | 1538 | 2800 | 0.0591 | 0.580 | 125170 | 28140 | 2150 | 4740 | 2.29 | 90.3 |
| 23 | 1.4 | 40 | 16.2 | 1538 | 2800 | 0.0581 | 0.570 | 125480 | 28208 | 2200 | 4860 | 2.34 | 92.1 |
| 8 | 1.6 | 20 | 9.1 | 1371 | 2500 | 0.0656 | 0.643 | 116980 | 26299 | 1870 | 4130 | 1.97 | 77.4 |
| 7 | 1.6 | 20 | 11.6 | 1538 | 2800 | 0.0657 | 0.644 | 119030 | 26758 | 1800 | 3965 | 2.01 | 79.2 |
| 9 | 1.6 | 30 | 11.2 | 1538 | 2800 | 0.0619 | 0.607 | 118760 | 26698 | 1760 | 3885 | 1.99 | 78.5 |
| 10 | 1.6 | 40 | 5.7 | 1204 | 2200 | 0.0624 | 0.612 | 114800 | 25807 | 2000 | 4410 | 1.89 | 74.6 |
| 11 | 1.6 | 40 | 8.2 | 1371 | 2500 | 0.0605 | 0.593 | 116950 | 26291 | 1950 | 4310 | 1.95 | 76.9 |
| 12 | 1.6 | 40 | 10.7 | 1538 | 2800 | 0.0600 | 0.588 | 118770 | 26700 | 1830 | 4030 | 1.99 | 78.4 |
| 13 | 1.6 | 40 | 13.9 | 1760 | 3200 | 0.0602 | 0.590 | 121130 | 27232 | 1820 | 4015 | 2.04 | 80.3 |
| 29 | 1.6 | 50 | 5.5 | 1204 | 2200 | 0.0615 | 0.603 | 114600 | 25764 | 2060 | 4540 | 1.91 | 75.2 |
| 30 | 1.6 | 50 | 8.2 | 1371 | 2500 | 0.0593 | 0.582 | 116490 | 26189 | 1980 | 4370 | 1.97 | 77.4 |
| 22 | 1.6 | 50 | 10.8 | 1538 | 2800 | 0.0587 | 0.576 | 118390 | 26615 | 1910 | 4200 | 2.01 | 79.2 |
| 31 | 1.6 | 50 | 14.0 | 1760 | 3200 | 0.0589 | 0.578 | 120540 | 27099 | 1920 | 4230 | 2.05 | 80.8 |
| 28 | 1.6 | 60 | 10.1 | 1538 | 2800 | 0.0583 | 0.572 | 117920 | 26510 | 1990 | 4390 | 2.00 | 78.6 |
| 32 | 1.7 | 40 | 9.5 | 1538 | 2800 | 0.0608 | 0.596 | 116720 | 26239 | 1730 | 3820 | 1.89 | 74.4 |
| 17 | 1.8 | 20 | 9.0 | 1538 | 2800 | 0.0679 | 0.666 | 114960 | 25844 | 1720 | 3790 | 1.82 | 71.6 |
| 18 | 1.8 | 30 | 8.9 | 1538 | 2800 | 0.0638 | 0.626 | 115050 | 25864 | 1720 | 3790 | 1.81 | 71.4 |
| 19 | 1.8 | 40 | 8.5 | 1538 | 2800 | 0.0617 | 0.605 | 114910 | 25833 | 1670 | 3690 | 1.81 | 71.2 |
| 20 | 1.8 | 46 | 8.3 | 1538 | 2800 | 0.0610 | 0.598 | 114820 | 25812 | 1670 | 3690 | 1.81 | 71.1 |
| 21 | 1.8 | 50 | 8.5 | 1538 | 2800 | 0.0604 | 0.592 | 114840 | 25816 | 1690 | 3720 | 1.82 | 71.6 |

* At 10.1 km (33,000 ft) altitude, Mn 0.83; excluding fan cowl drag

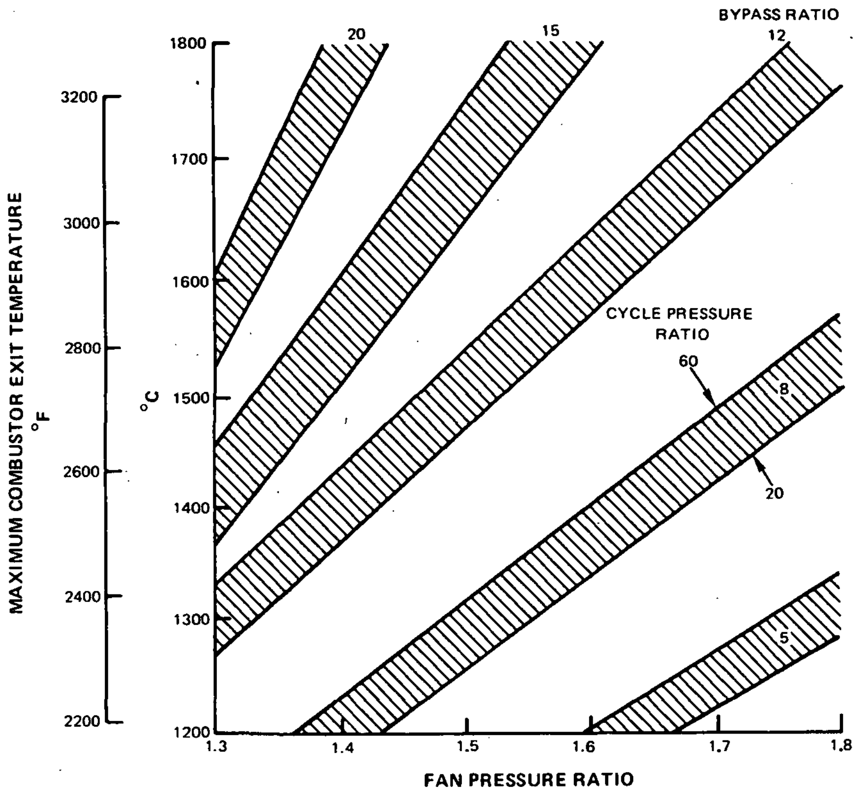



*Figure 5.2-1 Selected Cruise Design Bypass Ratio Level Trends*

The criteria for selecting engine thrust ratings were:

- Ratio of climb thrust to maximum cruise thrust = 1.10
- Ratio of sea level, 51.5-m/sec (100-knot) takeoff thrust to maximum cruise thrust = 3.45
- Maximum climb thrust flat rated to an International Standard Atmosphere (ISA) + 10°C (18°F) day
- Take-off thrust flat rated to an ISA + 17°C (31°F) day

These criteria provided a good balance to meet airplane thrust demands based on requirements of take-off field length, climb rate, and cruise speed for the study aircraft. The maximum combustor exit temperature occurred at the maximum climb rating at hot ISA +10°C (18°F) day conditions. Take-off combustor exit temperatures at ISA + 17°C (31°F) rated performance were lower than this climb combustor exit temperature. These criteria were consistent with current rating philosophies and airplane performance demands. The maximum climb level was used to define combustor exit temperature throughout the remainder of this report.

### 5.2.1 Fuel Savings Evaluation

The approach used to evaluate fuel savings consisted of first establishing trends for cruise TSFC. In establishing these trends, design and off-design performance characteristics were defined for all 26 cycles. Engine weight characteristics were then calculated for 5 selected cycles and used as a basis to define trends for the remaining 21 cycles. Finally, various airplane assumptions were made to assess the cycle trends on installed fuel consumption. The scope of this evaluation included the possible use of mixed exhaust systems and variable engine geometry.

#### 5.2.1.1 Cruise TSFC Trends

Trends in cruise TSFC were established based on the projected technological advances beyond the modern JT9D-59/70 high bypass ratio turbofan engine, which represents an advancement in itself beyond the JT9D-7 studied in the current engine evaluation. These trends indicate a total of 15 percent potential improvement in cruise TSFC (Figure 5.2.1.1-1). Approximately half of this improvement is attributed to projected improvements in component aerodynamics, while the remaining half is attributed to projected improvements in materials and cooling effectiveness. Improvements in structure-mechanics are also required to achieve the TSFC potential benefits. High mechanical rotor speeds enable gains in turbine adiabatic efficiency. Significant advances are necessary in engine main bearing technology to sustain these high operating speed levels.

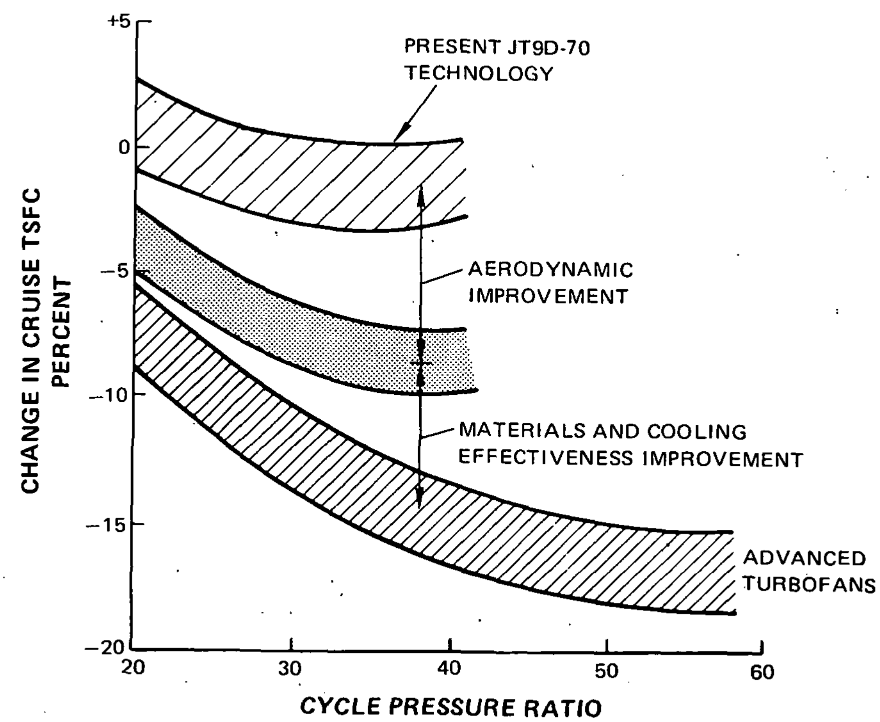



*Figure 5.2.1.1-1 Potential TSFC Improvement With Increasing Cycle Pressure Ratio*

Pressure ratio was found to be especially critical in achieving low TSFC for the advanced technology cycles. With current levels of materials and cooling technologies, increased cycle pressure ratio beyond present levels of 25:1 to 32:1 would require a sharp increase in the amount of air bleed for cooling these parts to acceptable temperature limits, causing a rapidly increasing loss in cycle efficiency and TSFC. With projected burner and turbine material properties improvements and higher cooling effectiveness, however, higher cycle pressure ratios can be exploited to achieve the large fuel consumption reduction indicated on Figure 5.2.1.1-1.

Bypass ratio levels higher than present were also found to be beneficial for improving TSFC of the advanced engines (Figure 5.2.1.1-2). Additional energy made available for propulsion by the more efficient engines was utilized to increase the low-pressure turbine specific power output to be absorbed by a higher bypass ratio fan. At bypass ratios higher than approximately 9, reduction gears in the low speed rotor were included between the low speed fan and high speed drive turbine to avoid excessive stages in the low-pressure turbine.

Increased turbine temperature, with projected materials advances, also had a beneficial effect on TSFC with a relatively small change in cooling air over a wide range of cruise temperature levels. Where a turbine temperature increase of 222°C (400°F) would require a total turbine cooling air increase of approximately 9 percent of engine airflow with present technology, a projected increase of less than 1 percent of engine airflow for turbine cooling would be required over the same temperature range with advanced technology. As a result, turbofan cycles with combustor exit temperatures over a 278°C (500°F) range provided cruise TSFC levels within 1 percent of the lowest level achieved at a maximum combustor exit temperature of 1538°C (2800°F).

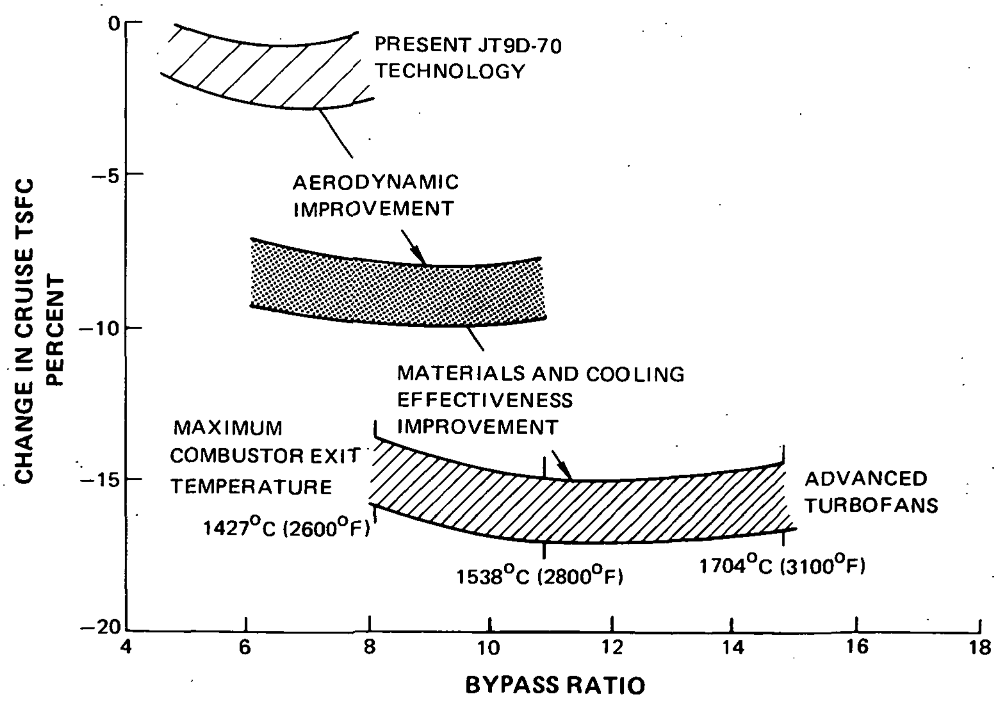



*Figure 5.2.1.1-2 Potential TSFC Improvement With Increasing Bypass Ratio*

5.2.1.2 Engine Weight Trends

Engine weight trends were estimated for the matrix of advanced technology turbofan cycles based on configurational and materials definition of five selected cycles (cycles 6, 9, 11, 12 and 13, Table 5.2-I).

Results of a weight trend evaluation, as related to cruise thrust, are shown in Figure 5.2.1.2-1. The thrust-to-weight ratios shown have been normalized to a level of 1.27, which can be considered typical of present day technology at the 10.1 km (33,000 ft), 0.83 Mach number, maximum cruise condition. The 1205°C (2200°F) weight trends are shown as approximate values since they represent extrapolations of data from the higher combustor exit temperatures. The weights include the engine, control, and accessories, exclusive of the nacelle. The procedure used to calculate nacelle geometry and weight is included in Appendix A. The advanced turbofan weights assume use of carbon epoxy in the fan blading and stators, and advanced titanium and nickel base alloys in the compressor and turbine sections.

The thrust-to-weight trends in Figure 5.2.1.2-1 can be envisioned as the result of the interaction between the effects of cycle variables on specific thrust capability (thrust per unit of total fan flow) and specific weight. Specific thrust capability, at a given bypass ratio, is significantly enhanced by increasing combustor exit temperature. Therefore, the thrust-to-weight ratio trends, at a fixed value of bypass ratio, primarily reflect the increase in specific thrust as combustor exit temperature is raised. At a given level of temperature, specific thrust typically decreases faster than weight as bypass ratio is increased. This accounts for the downward trends shown in the figure. Finally, increased compression system weight with higher cycle pressure ratios results in the attendant lower thrust-to-weight ratios of these cycles. At the very high pressure ratios, specific thrust also begins to decay.

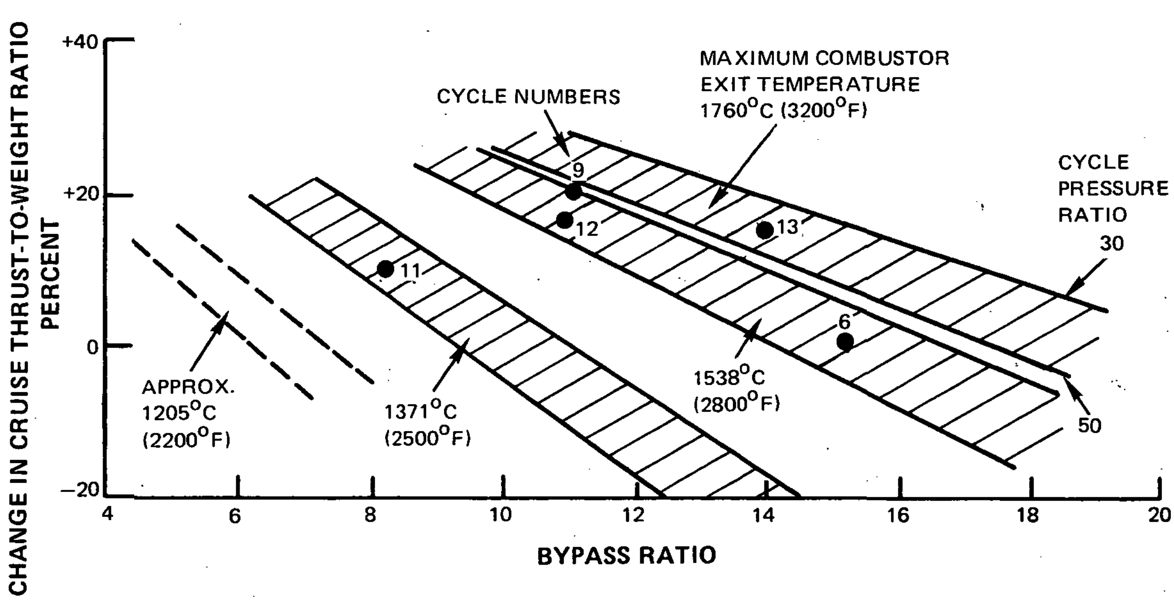



*Figure 5.2.1.2-1 Potential Turbofan Engine Weight Reduction with Advanced Technology*

The best combination of cycle parameters for highest thrust-to-weight ratio is not the same as the best combination of these parameters for lowest TSFC. The selection of the cycle parameters for low fuel consumption depends on the relative sensitivity to thrust-to-weight ratio and TSFC.

### 5.2.1.3 Airplane Configurations

A domestic trijet and an international quadjet airplane configuration were used in the evaluation of the advanced technology engines. General aircraft parameters are listed in Table 5.2.1.3-I. Aircraft characteristics in both cases include high aspect ratio wings, supercritical aerodynamics, and advanced lightweight composite structure technology. The wing geometry was varied as a function of cruise speed to minimize fuel use. The aerodynamic characteristics, structural weight characteristics, and economic evaluation ground rules are all detailed in Appendix A. The selected aircraft configurations and characteristics utilized are the results of data interchanges among NASA Lewis, Langley, and Ames Research Centers.

**TABLE 5.2.1.3-I**

**AIRCRAFT PARAMETERS FOR ADVANCED TURBOFAN EVALUATION**

| | Domestic Aircraft | International Aircraft |
|---|---|---|
| Design Cruise Mach No. | 0.7 to 0.9 | 0.7 to 0.9 |
| Design Range, km (n. mi) | 5560 (3000) | 10200 (5500) |
| Nominal Mission Range, km (n.mi.) | 1300 (700) | 3700 (2000) |
| Number of Passenger Seats | 200 | 200 |
| Number of Engines | 3 | 4 |
| Take-Off Gross Weight, kg (lbm) | 116,000 (255,000) | 132,000 (290,000) |
| Maximum Take-Off Field Length, m (ft) | 2440 (8000) | 3200 (10500) |
| Max. Approach Speed at Max. Landing Weight m/sec (knots) | 69.5 (135) | 72.0 (140) |
| Seat Pitch, First Class, m (in.) | 0.965 (38) | 0.965 (38) |
| Seat Pitch, Tourist, m (in.) | 0.864 (34) | 0.864 (34) |

### 5.2.1.4 Effect of Cycle Parameters on Airplane Fuel Consumption

The turbofan cycle parameters that provide the maximum potential for lowering fuel consumption were determined for both the domestic trijet and international quadjet aircraft configurations. The analysis included calculations of fuel burned at design range and on average flight distances for a typical airline operation. Engine performance estimates were made for the required flight conditions and power setting range of the aircraft for the twenty-six selected cycles which encompassed the study range. Each engine and airplane combination was sized to minimize the fuel requirement within the take-off distance and approach speed constraints.

Cycle Pressure Ratio – The effects of varying cycle pressure ratio on average stage length fuel consumption for the two aircraft are shown in Figure 5.2.1.4-1 for a given level of fan pressure ratio (1.6) and a maximum combustor exit temperature of 1538°C (2800°F) over a cruise Mach number range of 0.75 to 0.85. The trend of decreasing fuel burned with increasing pressure ratio reflects the lower TSFC potential predominating over the increasing engine weight. Weight accounts for only a twentieth of the effect on fuel use as does an equal percent change in TSFC for the systems evaluated. The predominance of the TSFC influence is shown in the figure for a 0.8 Mach number domestic trijet: The incremental effects of TSFC with weight held constant and weight with TSFC held constant are superimposed as functions of cycle variables. TSFC has an even more direct impact on the longer range international quadjet, so that the best overall pressure ratio shifts to a higher level. At cycle pressure ratios of about 50 to 60:1, however, additional gains in TSFC are not significant enough to offset the weight increase with further increase in cycle pressure ratio. This desirability for higher cycle pressure ratios to reduce fuel consumption does not change over the range of possible cruise Mach numbers because of the similar trends of TSFC with overall pressure ratio obtained at each cruise speed.

Maximum Combustor Exit Temperature – Increasing the turbine (combustor exit) temperature beyond present day levels, 1316°C (2400°F), to 1566°C (2850°F) with advanced technology resulted in a 2.5 percent reduction in fuel consumption for both the domestic trijet and international quadjet application, as shown in Figure 5.2.1.4-2. Although cruise TSFC is minimized at 1538°C (2800°F), the engine weight reduction and attendant lighter airplane possible with higher temperature levels result in the slightly higher temperature for minimum fuel. The choice of cruise Mach number clearly does not affect the desired temperature level at which fuel consumption is lowest. In addition, for the 1.6 fan pressure ratio, maximum combustor exit temperatures from 1427°C to 1704°C (2600°F to 3100°F) provide fuel consumption levels within 1.0 percent of the minimum.

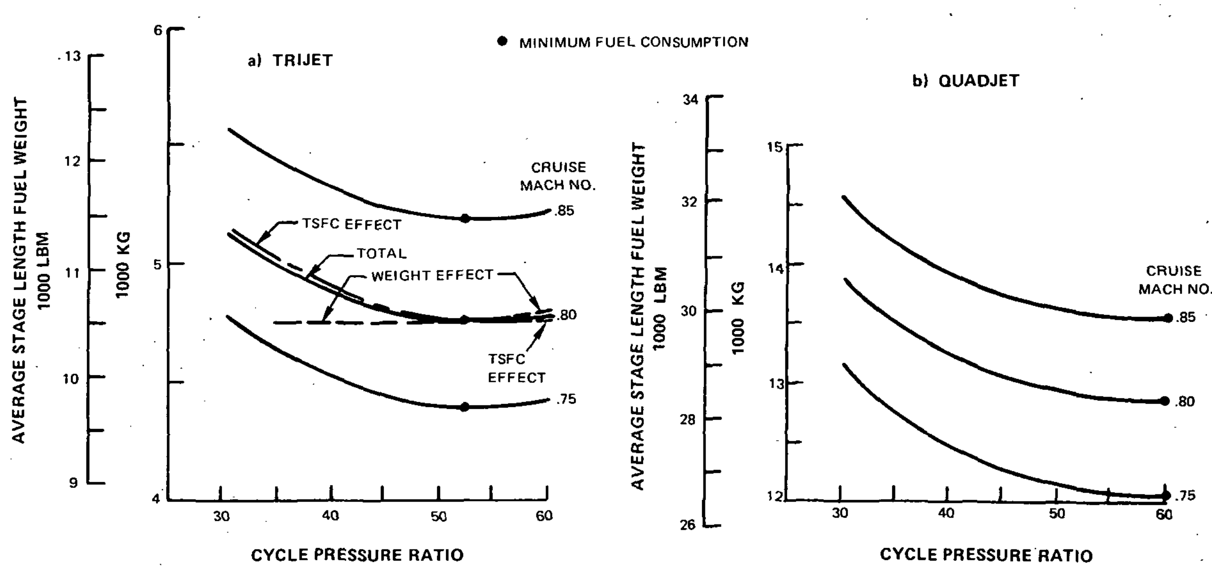



*Figure 5.2.1.4-1 Effect of Cycle Pressure Ratio on Fuel Consumption; Fan Pressure Ratio = 1.6:1, Maximum Combustor Exit Temperature = 1538°C (2800°F)*

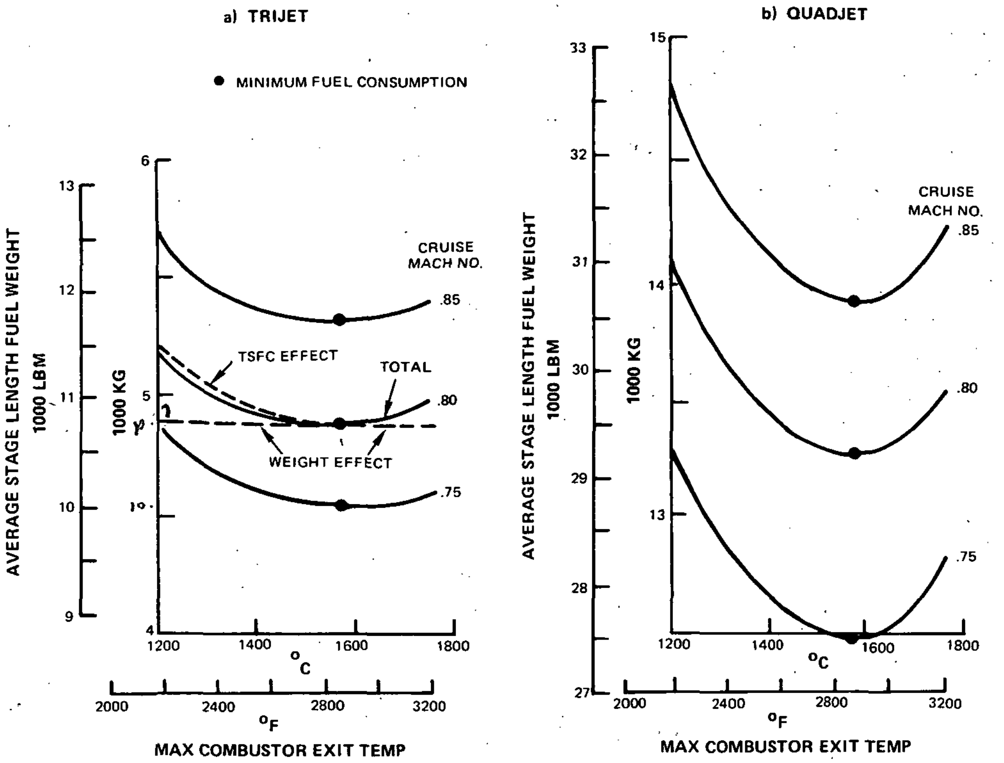



*Figure 5.2.1.4-2 Effect of Maximum Combustor Exit Temperature on Fuel Consumption; Fan Pressure Ratio = 1.6:1, Cycle Pressure Ratio = 40:1*

<u>Fan Pressure Ratio</u> – Variations in fan pressure ratio and, correspondingly, bypass ratio over the range of cruise speeds is presented in Figure 5.2.1.4-3. The potential improvements in fuel consumption due to lower TSFC at lower fan pressure ratios (higher bypass ratios) were offset completely by increased propulsion system weight and resultant higher aircraft gross weight.

Unlike changes in combustor exit temperature and cycle pressure ratio, fan pressure ratio reduction significantly increased the engine and nacelle size required to achieve a given thrust level because of the large reduction in nozzle jet velocity associated with reductions in nozzle pressure. Both nacelle drag and weight became important factors in assessing the fuel consumption trends. The effects of these factors, in addition to engine TSFC and engine weight, are illustrated in Figure 5.2.1.4-3 for the 0.80 Mach number domestic trijet. It becomes obvious that it is primarily the nacelle drag trend which drives the optimum cycle toward the more compact high fan pressure ratio levels.

As cruise speed increased, there was a perceptible shift in fan pressure ratio to higher levels for absolute minimization of fuel burned. For both the domestic trijet and the international quadjet, however, fan pressure ratios of from 1.55 to 1.70 provide fuel required levels within 1.0 percent of minimum over the cruise Mach number range of 0.75 to 0.85.

Therefore, as in the case of the cycle variables of cycle pressure ratio and combustor exit temperature, a single value of fan pressure ratio can provide essentially uncompromised fuel savings over a large cruise speed range.

These parametric studies have shown the progressively more stringent cycle requirements with advancing state-of-the-art. High pressure ratios and turbine temperature levels are required along with increased bypass ratios. The 1990's turbofan, as defined by these cycle requirements, represents a major challenge in aerodynamics, thermodynamics, and structural technology.

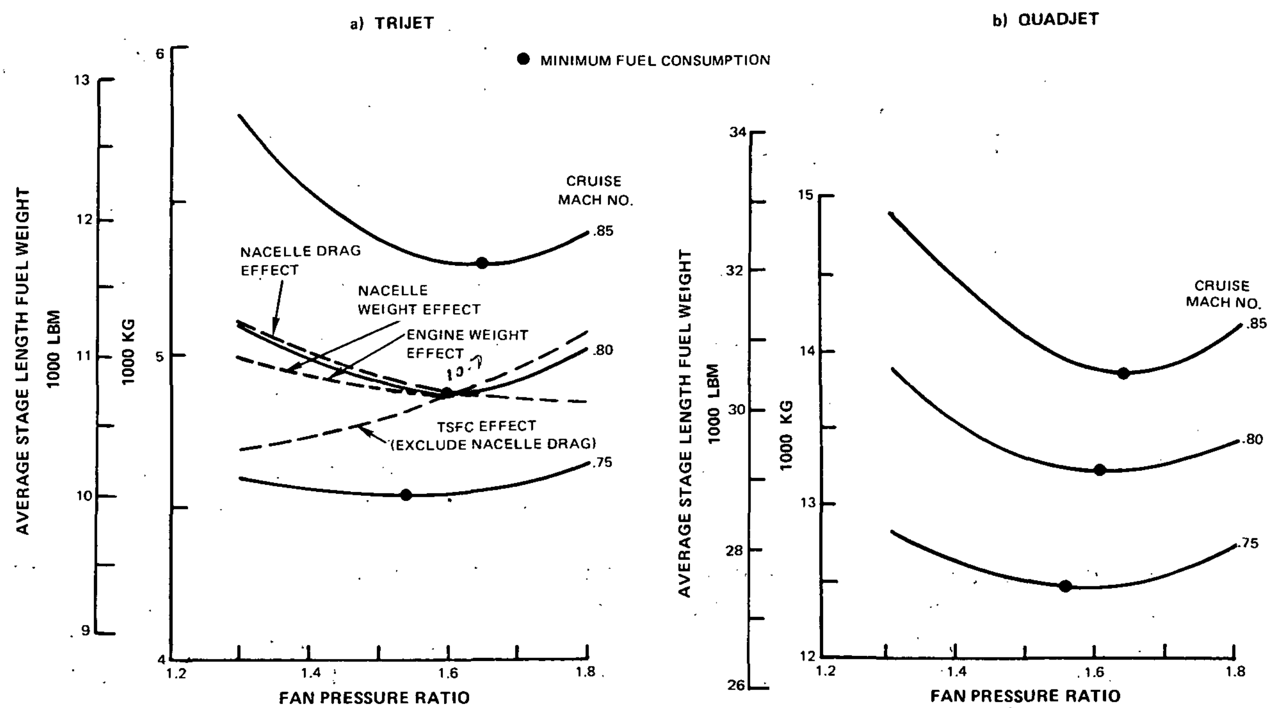



*Figure 5.2.1.4-3 Effect of Fan Pressure Ratio on Fuel Consumption; Cycle Pressure Ratio = 40:1, Maximum Combustor Exit Temperature = 1538°C (2800°F)*

### 5.2.1.5 Mixed Exhausts

An engine cycle was extracted from the parametric engine series for mixing analysis. The baseline cycle used for this study is shown as the first column of Table 5.2.1.5-I. As such, this cycle is not mixable because of the low pressure level of the turbine discharge as compared with the fan discharge air. The cycle was made more mixable by reducing the bypass ratio and increasing turbine discharge pressure into the mixable range. Mixed exhaust cycles were defined in this region for several combinations of mixer Mach numbers assuming a constant mixing efficiency of 75 percent.

TABLE 5.2.1.5-I

MIXED EXHAUST STUDY TURBOFAN CYCLES AND PERFORMANCE COMPARISON

| | Baseline Non-Mixed | Modified Non-Mixed | Mixed Exhaust |
|---|---|---|---|
| Cycle: | | | |
| Bypass Ratio | 10.8 | 9.6 | 9.6 |
| Fan Pressure Ratio | 1.6 | 1.6 | 1.6 |
| Overall Pressure Ratio | 40 | 40 | 40 |
| Max. Combustor Exit Temp., °C (°F) | 1538 (2800) | 1538(2800) | 1538(2800) |
| Mixer Inlet Mach Number, Engine Stream | – | – | 0.4 |
| Mixer Inlet Mach Number, Duct Stream | – | – | 0.5 |
| Performance: | | | |
| Relative TSFC Potential | 1.000 | 1.017 | 0.996* |
| $\Delta$TSFC – Duct Pressure Loss | 0 | 0 | +0.006 |
| $\Delta$TSFC – Nacelle Drag and Nozzle Thrust Coefficient, $C_v$ | 0 | 0 | +0.004 |
| Subtotal | 1.000 | 1.017 | 1.006 |
| $\Delta$TSFC Equivalent (Intallation Weight Change, Including Duct Thrust Reverser) | 0 | –0.003 | +0.001 |
| Relative Total Equivalent TSFC | 1.000 | 1.014 | 1.007 |

*75 Percent Mixing

From this array, a cycle was selected which, when drag and weight trends were estimated, minimized fuel consumption. Then both non-mixed and mixed exhaust versions of this cycle were defined for study. These two engines are summarized in the second and third columns of Table 5.2.1.5-I. Relative to the baseline non-mixed exhaust turbofan, reduction of bypass ratio to the mixable level increased TSFC 1.7 percent. Addition of a 75 percent efficient forced mixer to the mixable cycle could improve TSFC 2.1 percent excluding differences in internal duct pressure losses, nozzle thrust coefficients, or nacelle drag. Pressure loss, nozzle velocity coefficient, nacelle drag, and weight estimates were converted to TSFC changes using influence coefficients resulting in a 0.7 percent relative loss in equivalent TSFC for the mixed exhaust system.

A substantial 10 percent improvement in installed specific thrust was achieved at the lower bypass ratio of the mixed exhaust engine because of the increased thrust potential of the larger gas generator. At the same time, however, the nacelle surface area increased to the extent that, for the mixed exhaust engine, a net installed system weight increase of 2.5 percent was required. As a result, no potential for reducing fuel consumption was shown with the addition of a mixer. Consideration of other factors, such as jet noise and effects of detailed structural differences, were not included as part of this comparison and would require further study.

#### 5.2.1.6 Variable Engine Geometry

Several variable engine geometry features were screened in a preliminary manner to determine the potential for fuel consumption improvement. Variable-geometry stators in the compressor were included in all of the study engines, and, as such, are not reported separately here. The additional variable geometry features studied as discrete changes included: (1) variable pitch fan blading, (2) variable stage turbine vanes, (3) variable nozzles, and (4) combined variable turbine vanes and nozzles. These features are described in the following paragraphs. None of the features were found to offer a significant reduction in fuel consumption.

Variable Pitch Fan – The variable pitch feature in a fan can be used in several ways to provide improved engine or system performance. These include more optimum fan cruise performance, improved surge margin at sea level, and elimination of a thrust reverser.

In the analysis of this concept, the following ground rules were used: (1) fan design point efficiency is the same for fixed or variable pitch, and (2) fan off-design characteristics are the same for the variable pitch fan at nominal pitch as for the fixed pitch fan. The pitch-dependency characteristics were based on maps developed by Hamilton Standard Division of United Technologies Corporation (ref. 2).

A fan pressure ratio of 1.4 at the fan design point was selected for the variable pitch turbofan study as a reasonable compromise between higher fan pressure ratios, which the parametric study indicated would provide better airplane performance, and the lower fan pressure ratios, required to provide the operational advantage for the variable pitch feature.

Engines with low fan pressure ratios have a fan stability problem during sea level static operation due to the unchoking effect of the fan nozzle relative to cruise. In fixed pitch fans, this effect can be controlled by opening up a variable fan duct nozzle to reduce back pressure on the fan. This requires either a two-position nozzle, which results in an installation penalty, or a permanent lowering of the cruise fan operating line, which results in a large performance penalty. The variable pitch fan offers the possibility of increased fan stability through a fan pitch change at sea level.

Several cases were studied using the features of a variable pitch fan in several combinations. The base for the comparison was a fixed pitch fan with a two-position nozzle at sea level for fan surge margin control. One selected variable pitch fan used the variable pitch feature as a thrust reverser only. Cruise TSFC was the same as the fixed-pitch base since the variable pitch feature in a fan was assumed not to change design fan efficiency. As another case, the full potential of the variable pitch fan was assumed and the two-position nozzle and target thrust reverser were both eliminated. The variable pitch mechanism was therefore used both to provide take-off stability and serve as a reverser. The fan pitch was reduced at take-off by 0.26 radian (15°) and the fan speed increased by over 20 percent to accommodate nozzle back pressure without incurring surge. The weight penalty associated with the increased take-off speed was estimated to be greater than that associated with the alternative two-position duct nozzle. In addition, flared duct nozzle flaps were believed necessary for efficient reverse operation, where the nozzle serves as an air inlet. Therefore, the principal advantage of variable pitch was seen to lie in the elimination of the target thrust reverser.

On this basis, a comparison with the baseline fixed pitch fan was made. The variable pitch fan configuration consisted of long-chord, unshrouded fan blading with a harmonic drive variable pitch mechanism and a planetary single-stage reduction gear system. The fixed pitch fan was studied as a tip shrouded configuration. The root diameter of the variable pitch fan was 12 percent larger than the fixed pitch counterpart to accommodate the gearing volume requirements. Therefore, the tip diameter was two percent larger than the fixed pitch fan. The variable pitch fan blade-to-stator spacing was doubled to maintain the same spacing-to-chord ratio and fan noise level. The fan cowl dimensions were correspondingly increased with the fan changes. With these changes, the variable pitch fan configuration used as a reverser was estimated to be approximately 10 percent lighter than the fixed pitch system with a reverser cascade. Cowl drag increased by approximately 24 percent, however, because of the larger cowl. These differences, taken together, resulted in a 1.5 percent higher calculated fuel use estimate for the variable pitch fan in a 0.8 Mach number target. Therefore, it was concluded that the variable pitch fan did not offer any significant fuel savings potential.

Variable Geometry Turbine – Variable stagger vanes were studied in both the high-pressure and low-pressure turbines to vary the turbine flow capacity and modify the cycle characteristics during cruise operation.

Varying the pitch of the high-pressure turbine first vane had the singular cycle effect of altering the compressor pressure ratio. Closing the vane staggers caused back pressuring of the compressor, increasing compressor pressure ratio and reducing compressor surge margin. For example, a 5 percent reduction in turbine flow capacity cut the compressor surge margin in half during cruise operation. It is conceivable that the surge margin could be reduced by this amount since it includes allowances for engine power transients and inlet flow distortion which normally become critical factors at other than cruise conditions. The resulting increase in overall pressure ratio produced a 0.8 Mach number cruise TSFC potential improvement of slightly over 1 percent. This potential was reduced to less than 0.5 percent when a turbine efficiency penalty of one percentage point for the variable geometry features was taken into account. Based on test results to date, this penalty is believed to be extremely optimistic. Therefore, variable high-pressure turbine geometry was not recommended for further analysis.

Increasing the flow capacity of the low-pressure turbine by opening the vane stagger increased compressor corrected flow and pressure ratio at a rate corresponding to the changes along the standard compressor operating line. The resulting cycle changes consisted of a higher overall pressure ratio, which reduced TSFC, and reduced bypass ratio, which increased TSFC. At the cruise condition of 0.8 Mach number, increasing the turbine flow capacity by 7 percent provided the potential for less than a 0.5 percent TSFC improvement relative to the fixed geometry turbine. When the optimistic turbine efficiency penalty of one percentage point was taken into account, no TSFC improvement was estimated.

Variable Nozzle Geometry – Neither core nor fan duct nozzle geometry showed potential for significantly reducing engine fuel consumption. For engines with fan pressure ratios of 1.4 or lower, a two-position fan duct nozzle was required, however, to provide adequate sea level stability with best cruise performance. The use of variable fan duct nozzle geometry for cycles with higher pressure ratio did not significantly change performance without an unacceptable reduction in fan stability margin.

Primary nozzle variation was found to have a very secondary effect on cruise TSFC. TSFC varied only 0.2 percent for a 30 percent variation in thrust area.

Combined Variable Turbine and Nozzle Geometry – In order to explore the possibility of synergistic benefits of combined geometry variation not seen with individual changes, turbine and nozzle geometries were varied together as an additional study task.

It was stated above that some TSFC improvements could be made by controlling the operating lines of the fan and compressor, but usually at the cost of reduced stability margin. The next step is to use the turbine and jet area variation in combination to maintain the design operating lines. Improved cycle and propulsive efficiency can be achieved at reduced power by obtaining an optimum combination of fan and compressor match at any desired flight condition and power setting, without sacrificing stability margin, by jointly altering the turbine vane and exhaust nozzle flow capacities on the design operating lines. By comparing the match points of the fixed geometry cycle to match points of the variable geometry cycle giving optimum performance at two selected part power conditions, it was seen that the biggest shift was an increase in high-pressure compressor match. This means that the variable geometry cycle has a higher overall pressure ratio than the fixed geometry cycle at the power settings under consideration.

Table 5.2.1.6-I shows the TSFC improvement that is achieved through use of variable turbine and nozzle areas to obtain an optimum match at the two conditions of conventional cruise and cruise to an alternate field. When there is no turbine efficiency penalty for incorporating

**TABLE 5.2.1.6-I**

**EFFECT OF COMBINED ENGINE GEOMETRY VARIATION ON THRUST SPECIFIC FUEL CONSUMPTION**

**1.6 Fan Pressure Ratio**
**40 Cycle Pressure Ratio**
**1538°C (2800°F) Maximum Combustor Exit Temperature**

| | Typical Cruise | Cruise to Alternate Field |
|---|---|---|
| Altitude, km (ft) | 10.1 (33,000) | 6.1 (20,000) |
| Mach Number | 0.83 | 0.72 |
| Power Setting, Maximum Cruise | 0.85 | 0.60 |
| Flow Area Change From Nominal, % | | |
| High-Pressure Turbine | –6.1 | –8.1 |
| Low-Pressure Turbine | +14.3 | +23.8 |
| Primary Nozzle | 0 | –6 |
| Duct Nozzle | 0 | 0 |
| TSFC Change Relative to Nominal, % | | |
| Excluding Variable Geometry Efficiency Penalty | –1.1 | –2.0 |
| Including Variable Geometry Efficiency Penalty | +1.3 | +2.1 |

the variable geometry feature, the TSFC improvement was from 1.1 to 2.0 percent. When the turbine efficiencies were penalized for variable geometry (even for the least expected amount of penalty), TSFC of the variable geometry cycle was poorer than fixed geometry cycles by 1 to 2 percent. Therefore, the addition of combined variable turbine and nozzle geometry does not appear warranted.

### 5.2.2 Economic Evaluation

The economic evaluation of the advanced turbofan cycles consisted of determining trends for engine price, maintenance cost, and direct operating cost. Initially, the trends for engine price and maintenance cost were defined for the 5 selected cycles (cycles 6, 9, 11, 12, and 13 in Table 5.2-I) to provide a basis for estimating these characteristics for the remaining 21 cycles. This information, in combination with the previous TSFC and weight estimates, was then used to compute direct operating cost (DOC) and return on investment (ROI) for all 26 cycles.

The assumptions and procedures used for calculating DOC and ROI are presented in Appendix A. The most significant assumptions included fuel costs, 8 cents per liter (30 cents per gallon) for domestic fuel and 12 cents per liter (45 cents per gallon) for international fuel; maintenance direct labor of $7.30 per manhour; 55 percent revenue load factor; and a 15-year straight line airplane depreciation.

#### 5.2.2.1 Engine Price Trends

Generalized engine price trends were established for the purpose of evaluating the impact on airline operating economics. Engine price trends cannot be simply correlated with engine cycle parameters since engine price is also related to the specific component arrangement and turbomachinery stage designs. For example, price trends of direct-drive fan systems are extremely sensitive to the fan tip speed selection: the lower the tip speed the greater the number of stages in the low-pressure turbine and the higher the engine price. However, an increasing price trend was seen with higher bypass ratios and higher cycle pressure ratios. An increase in cycle pressure ratio of from 30:1 to 50:1 nominally increased price by 5 percent. Since the higher bypass ratios were achieved by utilizing higher levels of turbine temperatures and low rotor reduction gearing, the trends shown reflect these effects.

#### 5.2.2.2 Engine Maintenance Cost Trends

In order to assess the impact of these advanced fuel-conservative engines in relation to the airlines' operating costs, it was necessary to estimate the cost for maintaining the advanced engines. The elements comprising maintenance costs include both labor and material costs.

The physical configurations of the engines which held most promise of reducing fuel consumption were so similar that only shop maintenance cost was estimated (shop maintenance accounts for 90 percent of airline engine maintenance costs). The long form method of assessing engine maintenance costs (ref. 3) was used with modifications reflective of maintenance improvements projected for the 1990's.

Substantial improvements in maintenance technology were predicted for the engines. For example, it was predicted that the critical hot section part replacement rates will be equal to those of current, mature low-temperature engines with uncooled turbines. Significant advances in hot section part life and repairability are required to achieve these low parts replacement rates. The results of this study are shown in Figures 5.2.2.2-1 and 5.2.2.2-2. The five base cycles are located in the figures (see Table 5.2-I).

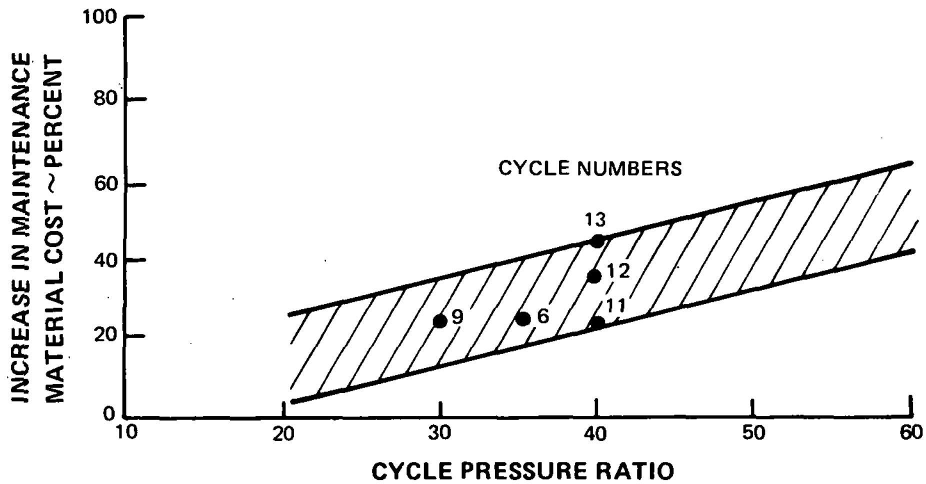



*Figure 5.2.2.2-1 Advanced Turbofan Maintenance Material Cost Trends*

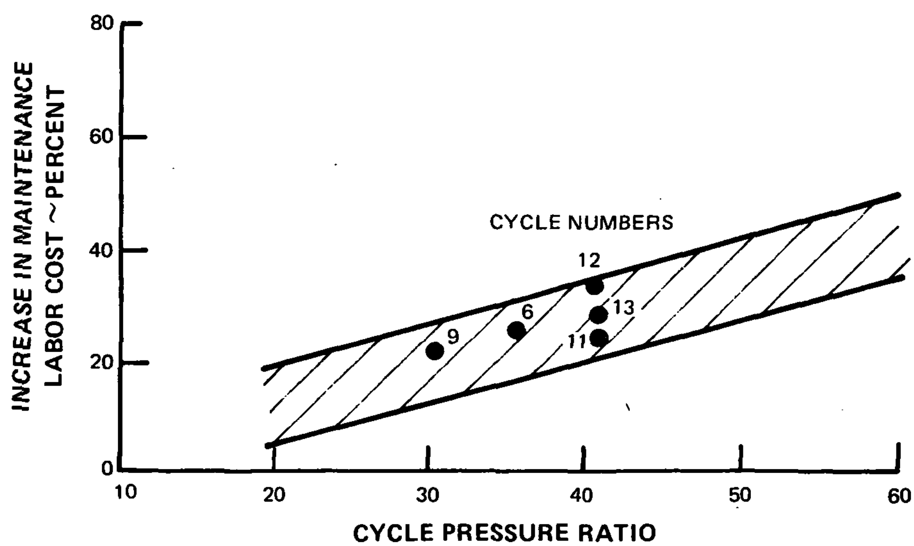



*Figure 5.2.2.2-2 Advanced Turbofan Maintenance Shop Labor Cost Trends*

5.2.2.3 Direct Operating Cost Trends

The effects of engine cycle changes on airline economics suggest that there are strong interactions among the major propulsion system economic variables of TSFC, weight, price, and maintenance cost. The effect of a straightforward change in one such variable, say TSFC, on airline economics is easily calculated. As TSFC changes, however, engine weight, price, and maintenance cost also change. It is the result of these interactions that produces the final airline economic picture: DOC and ROI. ROI trends generally follow DOC trends.

Effect of Cycle Pressure Ratio – The impact of engine economic variables and pressure ratio on DOC is shown in Figure 5.2.2.3-1. The net effect of all variables is indicated by the solid curves. The figure also shows the effect of each individual variable as dashed-line curves at Mach 0.8 for the domestic trijet. To obtain these curves the direct incremental cost of each variable was calculated as a function of cycle pressure ratio while other variables were held constant. These curves show that the net effect on cost is opposite to the effect of TSFC alone, due mostly to the large increase in maintenance material (replacement parts) costs as cycle pressure ratio is increased. The optimum cycle pressure ratio for DOC considerations is therefore at a significantly lower level than for fuel consumption considerations (see Figure 5.2.1.4-1).

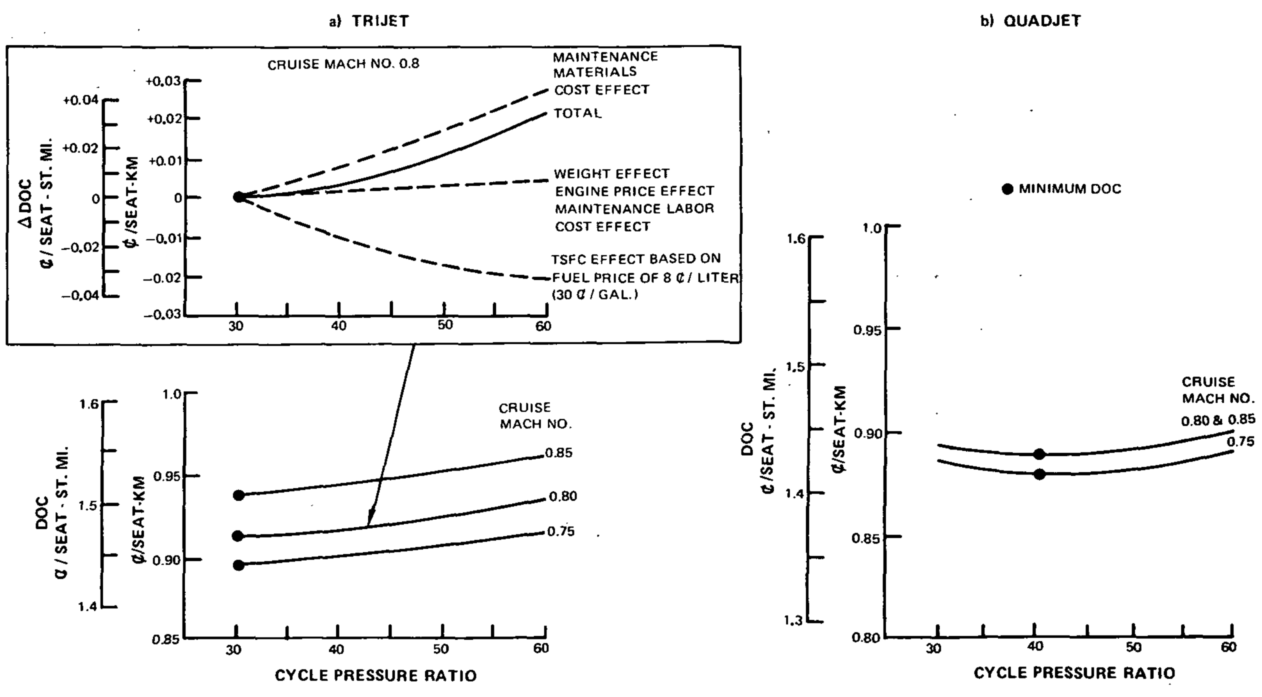



*Figure 5.2.2.3-1 Effect of Cycle Pressure Ratio on Direct Operating Cost; Fan Pressure Ratio = 1.6:1, Maximum Combustor Exit Temperature = 1538°C (2800°F)*

Effect of Combustor Exit Temperature – Direct operating cost trends with combustor exit temperature are shown in Figure 5.2.2.3-2. A discontinuity in the trends resulted from the step change in engine price during the switch-over from direct drive to geared fan systems as shown for the domestic trijet in the figure. The direct drive fan price trends used were for low speed optimum performance fans, where a steep price gradient versus bypass ratio was obtained (Section 5.2.2.1). The use of a higher fan speed to eliminate a turbine stage would considerably flatten the trend. However, even with the present discontinuity, DOC remained within 1.0 percent of constant over most of the temperature range. This result, as in the case of overall pressure ratio, again represented a trade between the counteracting influence of TSFC and maintenance material costs up to a combustor exit temperature of 1538°C (2800°F).

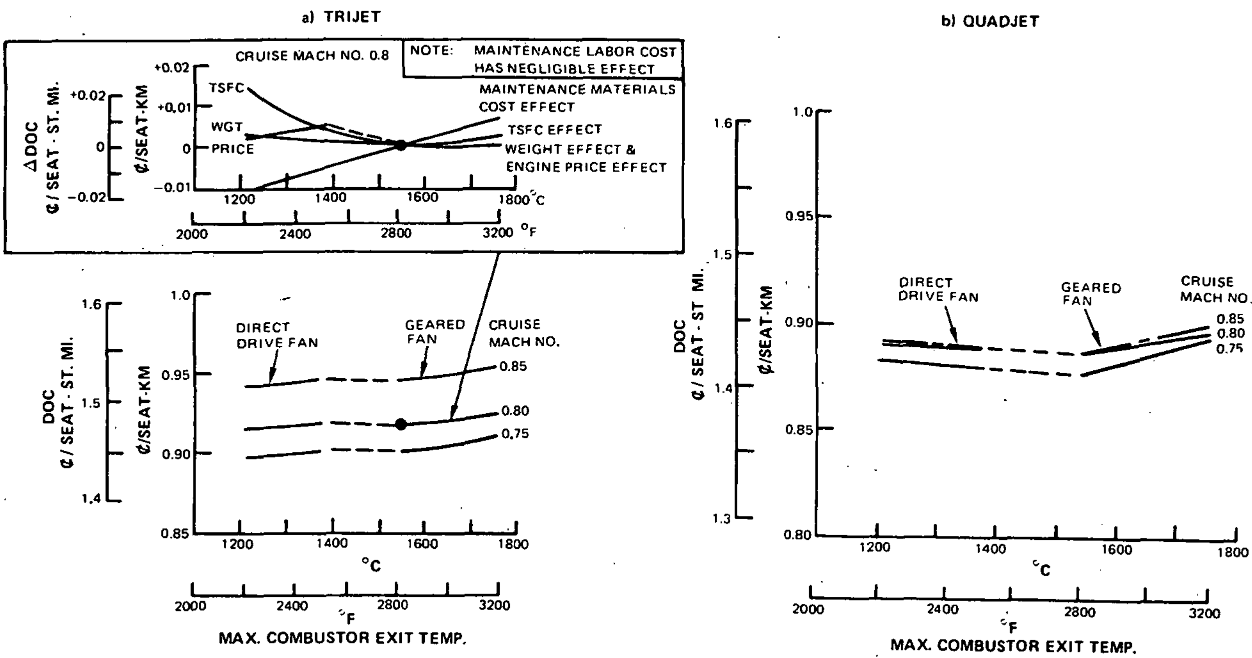



*Figure 5.2.2.3-2 Effect of Maximum Combustor Exit Temperature on Direct Operating Cost; Fan Pressure Ratio = 1.6:1, Cycle Pressure Ratio ≐ 40:1*

Effect of Fan Pressure Ratio – The effects of fan pressure ratio variation on DOC were a summation of the several variables affecting fuel consumption plus the effects of engine price, engine maintenance cost, and nacelle price. Trends are shown in Figure 5.2.2.3-3. A shift to slightly higher fan pressure ratios for minimum DOC than those for minimum block fuel was apparent at all cruise speeds (see Figure 5.2.1.4-3). This shift was caused by decreases in engine price and nacelle price with increasing fan pressure ratio. The nacelle price effect was due to the larger nacelle sizes required at lower fan pressure ratios. This shift was tempered somewhat by increases in engine maintenance cost with fan pressure ratio. The separate effects of the engine related variables are shown in Figure 5.2.2.3-3 for the domestic trijet at Mach 0.8.

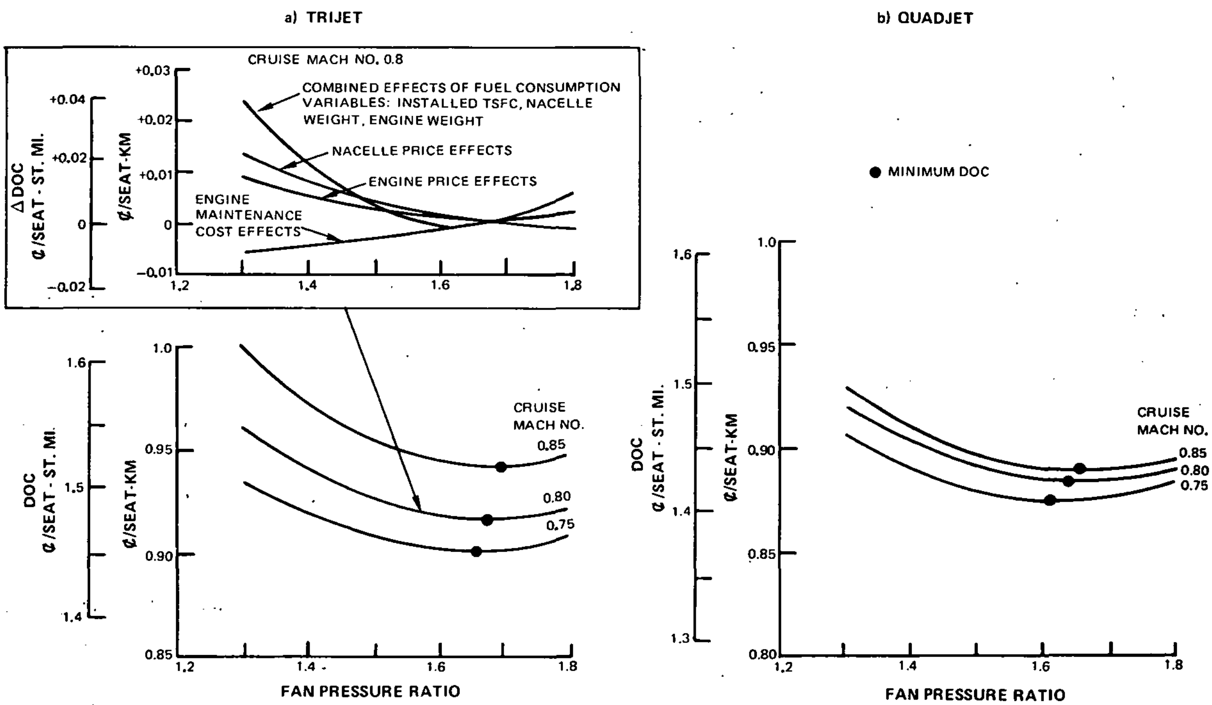



*Figure 5.2.2.3-3 Effect of Fan Pressure Ratio on Direct Operating Cost; Cycle Pressure Ratio = 40:1, Maximum Combustor Exit Temperature = 1538°C (2800°F)*

# 6.0 DISCUSSION OF RESULTS OF TASK III – REFINED ANALYSIS OF FUTURE ENGINES

The principal objectives of the Task III analytical studies were to select an engine cycle from the Task II parametric analysis and conduct a refined analysis of this cycle. A configurational screening study was conducted for four discrete engines leading to selection of a turbofan engine given the P&WA study designation STF 477. For this engine, both component conceptual aerodynamics and a mechanical design arrangement were defined. Acoustic nacelle contours, aimed at a FAR 36 minus 10 EPNdB noise capability, were also estimated. Engine performance, airplane performance, and airline economics were studied and compared with a synthesized 1975 technology turbofan cycle.

## 6.1 ENGINE CYCLE SELECTION

As the first step in the refined analysis of Task III, major engine cycle parameters which offered the greatest potential for reducing fuel consumption were selected. Desired levels of cycle pressure ratio, combustor exit temperature, and fan pressure ratio were selected, based on the results acquired from the Task II parametric evaluation.

### 6.1.1 Cycle Pressure Ratio

The Task II study established the need for high pressure ratios for low energy consumption. That study showed that a pressure ratio of 52:1 for the domestic trijet and 60:1 for the international quadjet provided the minimum fuel usage. However, with increasing pressure ratio, flow passage heights become smaller for the same mass flow rate, and it becomes increasingly difficult to maintain close running clearances. In addition, the potential exists for higher leakage through flanges and seals. On the basis of these considerations as well as maintenance cost trends, a maximum pressure ratio level of 45:1 was selected as an aggressive target.

Emissions estimates were made for the selected cycle pressure ratio advanced technology turbofan. These estimates were based on the current on-going emissions reduction programs for the JT8D and JT9D engines in addition to the NASA Experimental Clean Combustor Program (ECCP). The selected burner concept consisted of a swirl burner called a vortex burning and mixing (Vorbix) burner based on the ECCP design in combination with a modified pilot to improve low power emissions. This selection was based on the observed low emission levels of the Vorbix burner at intermediate and high engine power settings and the low carbon monoxide (CO) and total hydrocarbons (THC) emission characteristics of the aerating nozzles at low power settings.

The calculated emission levels are compared with representative JT9D production levels and proposed EPA standards in Figure 6.1.1-1 (ref. 4). The high pressure ratio of the fuel conservative turbofan aggravates the oxides of nitrogen ($NO_X$) generation at high power levels so that, even with emission advances, it is estimated that the EPA standards will be exceeded. Further advances in emissions technology is therefore required to meet the needs of the fuel conservative engine.

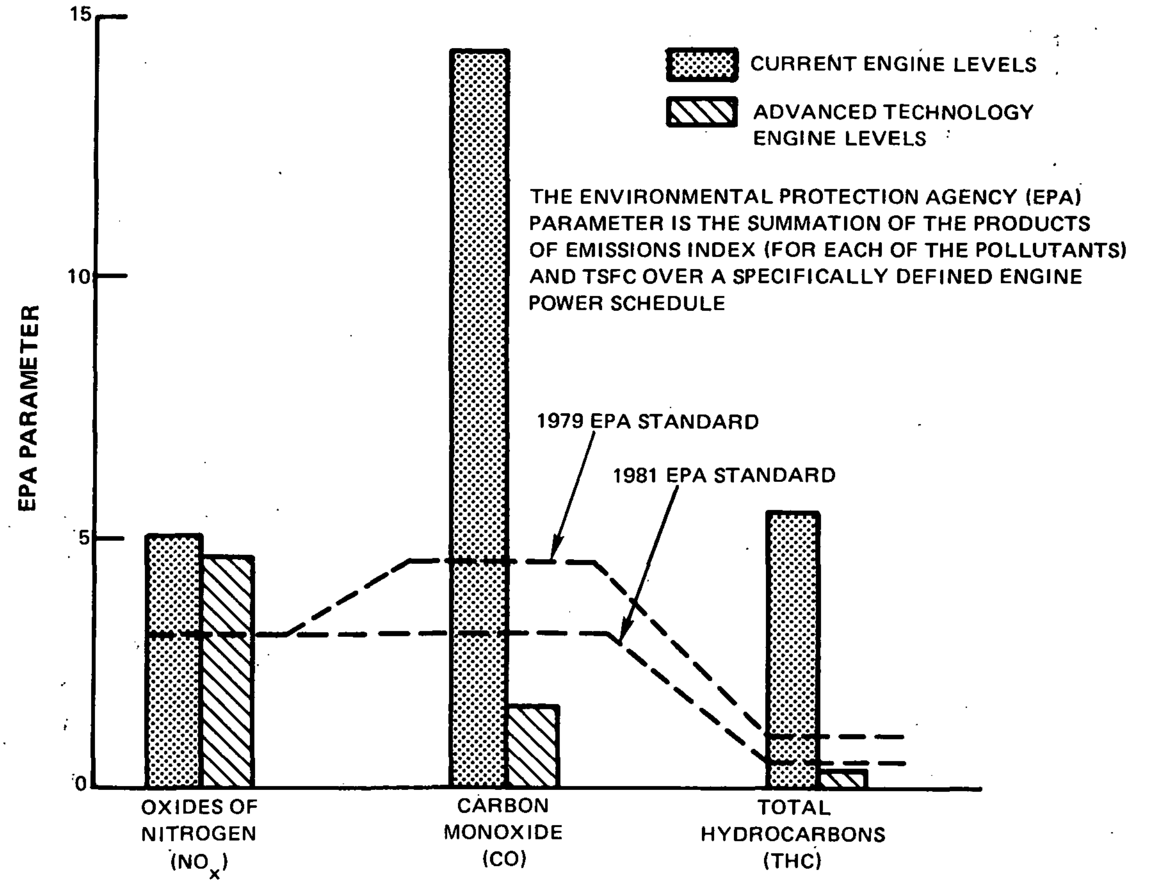



*Figure 6.1.1-1 Comparison of Present and Future Predicted Exhaust Emissions Levels*

### 6.1.2 Fan Pressure Ratio and Combustor Exit Temperature

The sensitivity of fuel usage to the cycle parameters, as determined in the parametric evaluation, is illustrated in Figure 6.1.2-1, with lines of constant direct operating costs superimposed as an economic trend indicator. The trijet trends shown typify the results for the quadjet analysis as well. In combination, fuel usage and operating cost trends represent compatible cycle requirements in that both are minimized over an overlapping range of cycle parameters.

Because the parametric results of Task II did not clearly define a best cycle, four engine cycles in the region of interest were selected (Table 6.1.2-I and Figure 6.1.2-1) and screened to provide the basis for final cycle selection. These cycles were evaluated as discrete data points for fuel consumption and economic evaluations. This led to the selection of Cycle B for refined analysis. This engine, given the P&WA study designation STF 477, is then used as the comparison base for the other engine cycles.

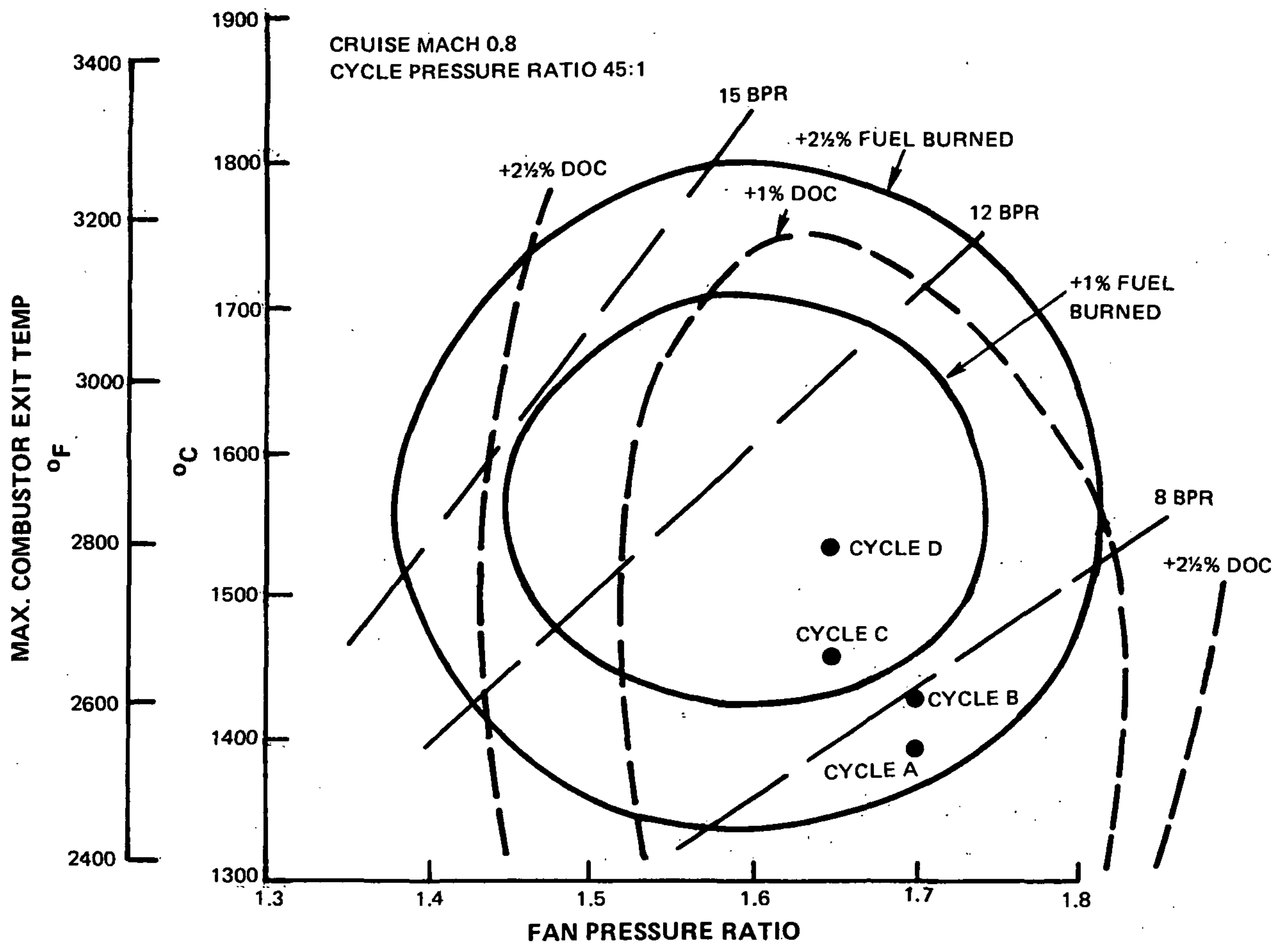



*Figure 6.1.2-1 Domestic Trijet Fuel Consumption and Direct Operating Cost Trends*

**TABLE 6.1.2-I**

**TURBOFAN CYCLES SELECTED FOR SCREENING FOR REFINED ANALYSIS**

| | Cycle A | Cycle B* | Cycle C | Cycle D |
|---|---|---|---|---|
| Cruise Point Design | | | | |
| Cycle Pressure Ratio | 45:1 | 45:1 | 45:1 | 45:1 |
| Fan Pressure Ratio | 1.70 | 1.70 | 1.65 | 1.65 |
| Max. Combustor Exit Temp., °C (°F) | 1399 (2550) | 1427 (2600) | 1454 (2650) | 1538 (2800) |
| Bypass Ratio | 7.6 | 8.0 | 8.9 | 10.1 |

*Cycle selected for refined analysis and given P&WA study designation STF 477.

## 6.2 ENGINE CONFIGURATIONAL SCREENING

For the engine cycles selected for screening, various possibilities exist regarding the arrangement of turbomachinery. Compression could be accomplished by means of a fan and high-pressure compressor combination, or a fan/low compressor/high compressor combination on two or three shafts. For engines with higher bypass ratios such as Cycle D a speed reduction gear is needed between the fan and the low-pressure turbine. Also, the division of compression split affects the turbine aerodynamics through the work balance requirement. As a result, a wide range of possible arrangements can be conceived. A comprehensive examination of all possibilities would require far greater analysis than could be conducted in this program. However, as summarized below, a preliminary look was taken at the configurational possibilities to assess the influence of major changes in arrangement of turbomachinery.

### 6.2.1 Compressor and Turbine Spool Arrangements

Configurational options available to the designer are illustrated in Figure 6.2.1-1 for fuel conservative turbofan cycle characteristics. In order to configure a high cycle pressure ratio engine without a low or intermediate compressor, high compressor pressure ratios of 28:1, or greater, are required – higher by more than 50 percent than any aircraft engine compressor known today. Because of the likelihood of intrastage mismatch and instability associated with very high pressure ratio compressors, the fan plus high compressor configurations were not considered further. For the configurations studied, a limiting pressure ratio of 18:1 was judged to be reasonable from a performance and stability viewpoint. Also, this pressure ratio provides a high enough work absorption requirement so that the low-pressure turbine vanes or blades do not require cooling and resultant aerothermodynamic penalties.

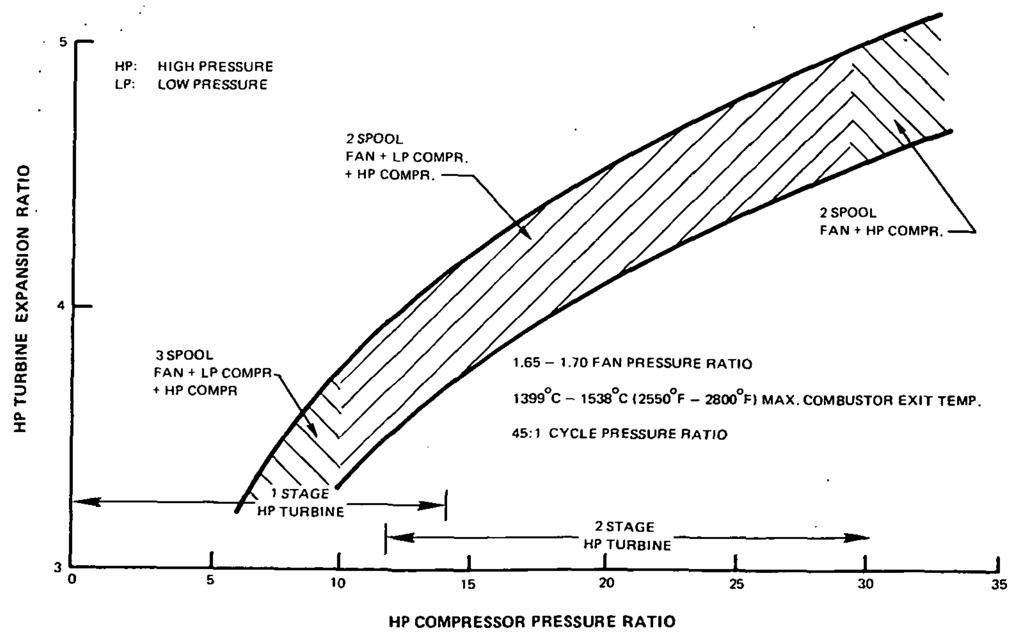
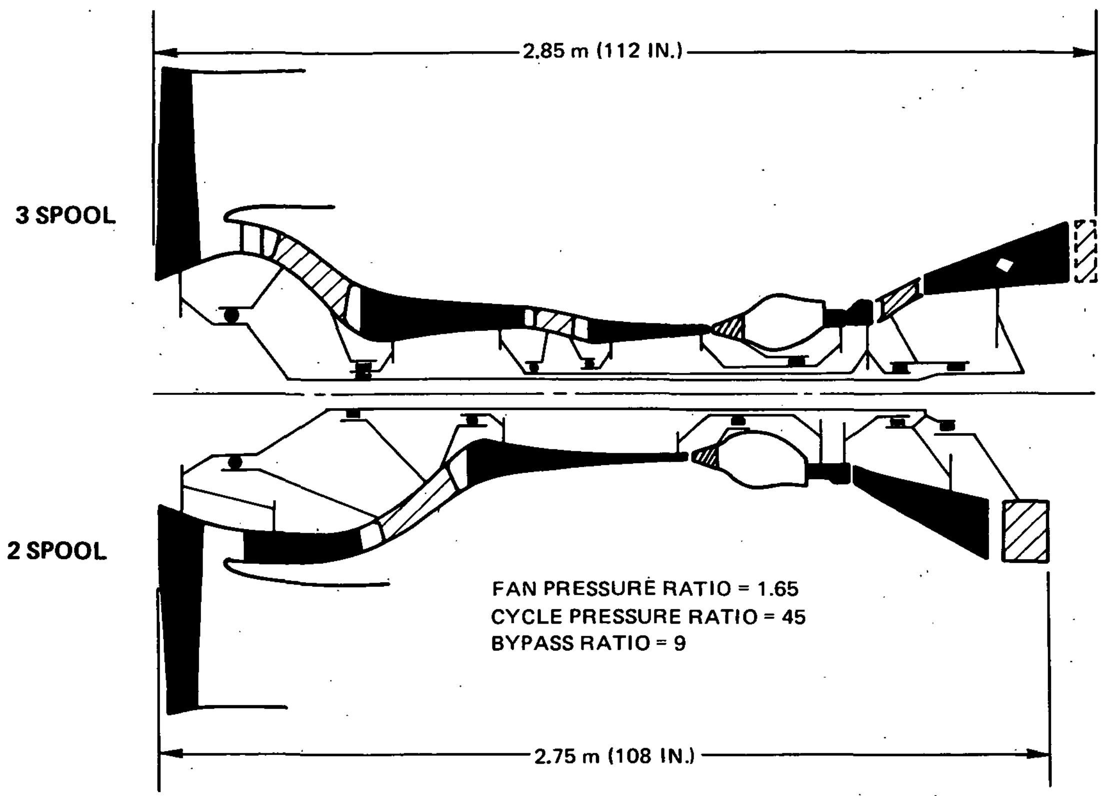



*Figure 6.2.1-1 Possible Engine Configurations for Fuel Conservative Engines*

The ability of a three spool compression system to achieve a higher average wheel speed provides the potential for minimizing the number of stages and amount of variable compressor geometry. The high pressure ratio of the fuel conservative engines could be especially responsive to the possible compression system improvements. Therefore, a preliminary comparison of a three spool and two spool, fan/low compressor/high compressor configurational comparison was made for the 45:1 pressure ratio cycle, as illustrated in Figure 6.2.1-2. It is estimated that the two configurations would have essentially identical performance capability. The potential cost and weight savings of the three spool design, which has fewer compressor stages and one less variable stator row, are countered by its additional shaft, two




| | 2 SPOOL | 3 SPOOL |
|---|---|---|
| FAN STAGES | 1 | 1 |
| COMPRESSOR STAGES | 13 (3 + 10) | 10 (4 + 6) |
| TURBINE STAGES | 7 | 7 |
| VARIABLE GEOM. COMPR. STAGES | 5 | 4 |
| MAIN BEARINGS | 6 | 8 |
| BEARING SUPPORTS | 3 | 4 |
| RELATIVE HIGH ROTOR SPEED | 1.0 | 1.22 |
| ENGINE LENGTH, m (IN.) | 2.75 (108) | 2.85 (112) |

*Figure 6.2.1-2 Two Spool Versus Three Spool Configuration Comparison of Fuel Conservative Turbofan Engines*

additional main bearings, and additional main bearing support system. Very small differences in either weight or cost would therefore be expected between the two configurations. Detailed design studies would be required to adequately define these differences. Because of the anticipated similarity in gross engine characteristics and concern over running three coaxial shafts in the small thrust size of the engines included in this study, the two spool, fan/low compressor/high compressor configuration type was selected as the preferred approach.

## 6.2.2 Final Configuration Selection

Each of the four cycles selected for screening were configured as two spool, fan/low compressor/high compressor (F-L-H) arrangements. A two stage high-pressure turbine was selected for the STF 477. Engine Cycle B was also studied with a single stage high-pressure turbine (Cycle B-1), with appropriate redistribution of compression between the low and high-pressure compressors. Relative cruise TSFC, engine weight, and engine price estimates were made for each of the five cases. Trade factors were then used to provide comparisons of fuel consumption and airline economics.

The results of this study, summarized in Table 6.2.2-I, illustrate the very small differences in the findings for all of the cycles and configurations studied.

The direct fan drive was selected over the geared fan system, even though the latter was indicated to offer a slightly lower fuel requirement. The uncertainty associated with a gear system capable of delivery of 18,600 kw (25,000 horsepower) with the 99 percent gear efficiency assumed (a 1 percent gear efficiency reduction would increase fuel burned by approximately 0.8 percent), in addition to its unknown maintenance costs and reliability, were primary considerations in selecting the direct drive system. Further detailed study is required to fully evaluate the complexities of the geared drive versus its potential for lower fuel consumption. Therefore, a gear technology program is recommended in Section 7.0 of this report.

**TABLE 6.2.2-I**

**TURBOFAN CONFIGURATIONAL SCREENING RESULTS**

| | Cycle A | Cycle B-1 | Cycle B-2 (STF 477) | Cycle C | Cycle D |
|---|---|---|---|---|---|
| **Fan Drive** | Direct Drive | Direct Drive | Direct Drive | Direct Drive | Geared |
| **Fan/Compressor Arrangement** | F-L-H | F-L-H | F-L-H | F-L-H | F-L-H |
| Compressor/Turbine Stages | 1-3-10/2-5 | 1-5-9/1-5 | 1-3-10/2-5 | 1-3-10/2-5 | 1-3-10/2-4 |
| Relative Cruise TSFC | 1.002 | 1.020 | 1.000 | 0.998 | 0.996 |
| Relative Weight | 0.992 | 1.064 | 1.000 | 1.086 | 1.080 |
| Relative Price | 0.992 | 0.998 | 1.000 | 1.032 | 0.998 |
| Relative Fuel Burned | 1.001 | 1.027 | 1.000 | 1.002 | 0.999 |
| Relative DOC | 1.000 | 1.008 | 1.000 | 1.004 | 1.001 |

## 6.3 PARAMETERS AND CHARACTERISTICS OF THE STF 477

### 6.3.1 STF 477 Component and Mechanical Description

The STF 477 advanced technology turbofan consists of a high-speed, single stage 1.7 pressure ratio fan, a three stage low-pressure compressor with a pressure ratio of 1.53, and an 18.2:1 pressure ratio high-pressure compressor in ten stages. A low emissions, two stage Vorbix combustor with aerating pilot nozzles is included to provide a 1427°C (2600°F) maximum average combustor exit temperature. The compression system is powered by a two stage, cooled high-pressure turbine and a five stage low-pressure turbine.

STF 477 parameters are summarized in Table 6.3.1-I. The engine was scaled to meet requirements.

A cross section of the STF 477 engine configuration is presented in Figure 6.3.1-1. The two rotor systems are supported by six bearings: three bearings supporting the low-pressure rotor and three bearings supporting the high-pressure rotor, one of which is an intershaft bearing at the rear of the engine. A bearing at the mid-engine location is included to provide additional support to minimize rotor deflections and help to minimize running clearances in the rear of the compressor and in the high-pressure turbine.

### 6.3.2 Advances in Technology Required for STF 477

The component characteristics of the STF 477 cycle were compared with those of a synthesized 1975 turbofan with JT9D-70 technology, and the resultant technology advancements are summarized in Table 6.3.2-I. This table shows the need for advancements over a broad spectrum in order to achieve the potential 1985 technology levels necessary for substantial fuel consumption reduction.

Based on the improvements summarized in Table 6.3.2-I, a comparison of the overall engine characteristics of the 1975 and 1985 technology turbofans shows the advanced turbofan has a 12.5 percent lower installed cruise TSFC potential if all advances are achieved. With the incorporation of advanced composite materials in the fan section and advanced titanium base and nickel base alloys in the compressor and turbine, an engine weight reduction of 22 percent was estimated.

### 6.3.3 Description of the Acoustic Nacelle for the STF 477

A definition of the STF 477 nacelle contours was based on the aerodynamic parameters of the common nacelle system designed for the JT9D engine. The contours and design parameters are shown in Figure 6.3.3-1. Inlet and duct lengths were established based on treated surfaces required to comply with a FAR 36 minus 10 EPNdB noise goal. The inlet diameter and contours were based on the aerodynamic considerations of pressure recovery and inlet drag at cruise and take-off conditions. The nacelle afterbody mean angle was set at 0.23 radian (13°) to minimize nacelle weight and improve performance, while remaining within the aerodynamic constraints determined by model testing.

TABLE 6.3.1-I

STF 477 ENGINE PARAMETERS

**PARAMETRIC DESCRIPTION**

| | |
|---|---|
| Base Size, Thrust, N (lbf)* | 118100 (26550) |
| Scaling Range, Thrust, N (lbf)* | 71200-178000 (16000-40000) |
| Nominal Cycle | |
| Fan Pressure Ratio | 1.70:1 |
| Bypass Ratio | 8.0:1 |
| Overall Pressure Ratio | 45:1 |
| Maximum Combustor Exit Temperature, °C (°F) | 1427 (2600) |
| Inlet Flow (corrected), kg/sec (lbm/sec) | 472 (1040) |
| Acoustics (Engine Plus Nacelle) | FAR 36 minus 10 EPNdB |

**PERFORMANCE (Representative Conditions)**

| Condition | Altitude | | Mach No. | Net Thrust | | TSFC | |
|---|---|---|---|---|---|---|---|
| | km | (ft) | | N | (lbf) | kg/hr/N | (lbm/hr/lbf) |
| Take-off** | 0 | (0) | 0 | 111610 | (25091) | 0.0290 | (0.284) |
| Max. Climb*** | 9.1 | (30000) | 0.8 | 32912 | (7399) | 0.0588 | (0.577) |
| Max. Cruise*** | 9.1 | (30000) | 0.8 | 29910 | (6724) | 0.0586 | (0.575) |

**WEIGHTS AND DIMENSIONS**

| | |
|---|---|
| Base Engine Weight, kg (lbm) | 1787 (3940) |
| Dimensions | |
| Maximum Diameter, m (in.) | 1.92 (75.6) |
| Overall Length, m (in.) | 2.88 (113.2) |
| Nozzle Throat Areas | |
| Duct, $m^2$ ($in.^2$) | 1.150 (1783) |
| Primary, $m^2$ ($in.^2$) | 0.303 (470) |

*Sea level take-off, 28.9°C (84°F) ambient temperature.

**Estimated performance calculated on basis of: U. S. Standard Atmosphere, 1962; 100 percent ram recovery; 1.04 kg/sec (2.3 lbm/sec) mid-compressor bleed; 1.01 kg/sec (2.4 lbm/sec) duct bleed; 112 kw (150 hp) extraction; standard day.

***Same conditions as take-off except bleed: 0.91 kg/sec (2.0 lbm/sec) mid-compressor; 0.45 kg/sec (1.0 lbm/sec) duct bleed.

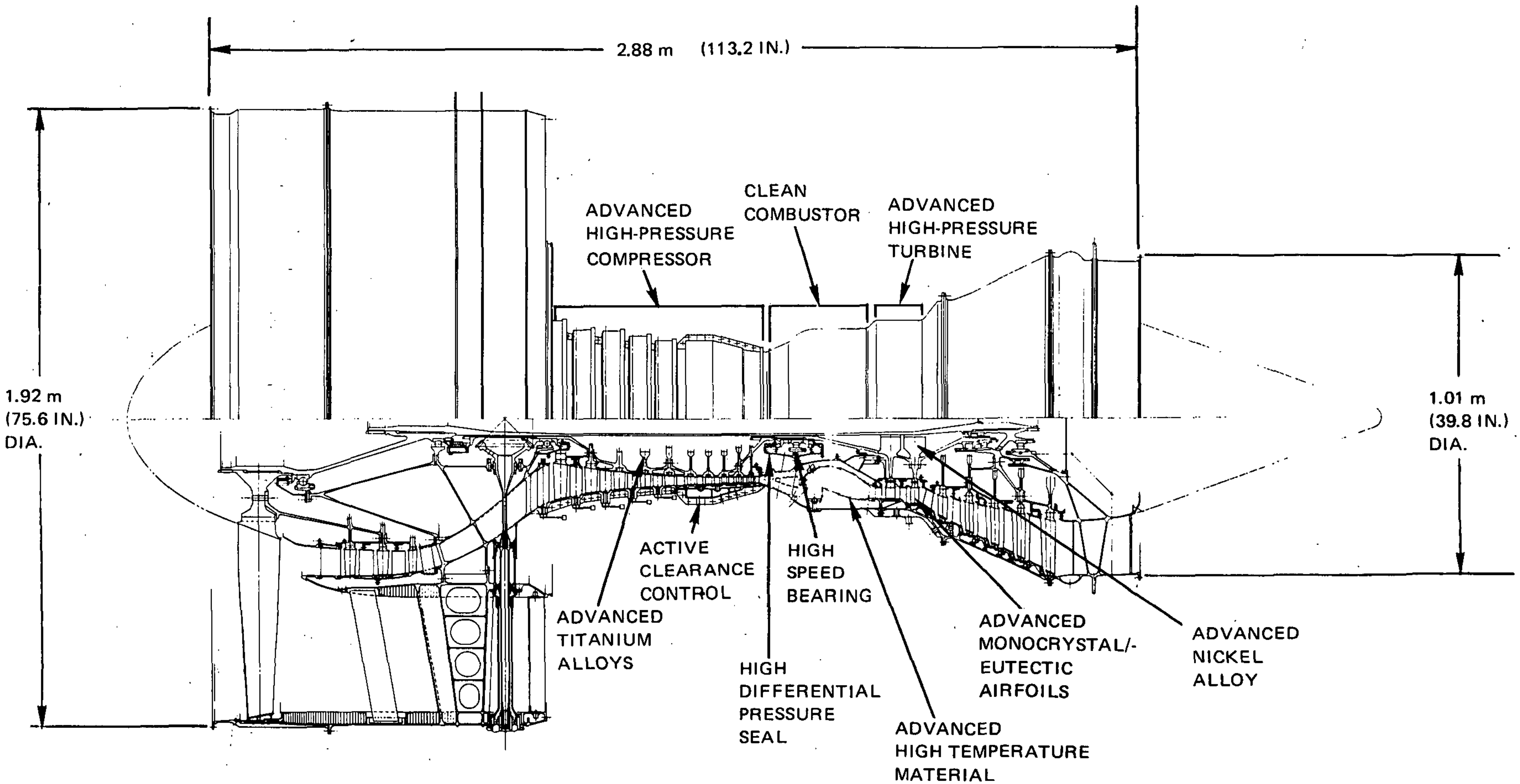
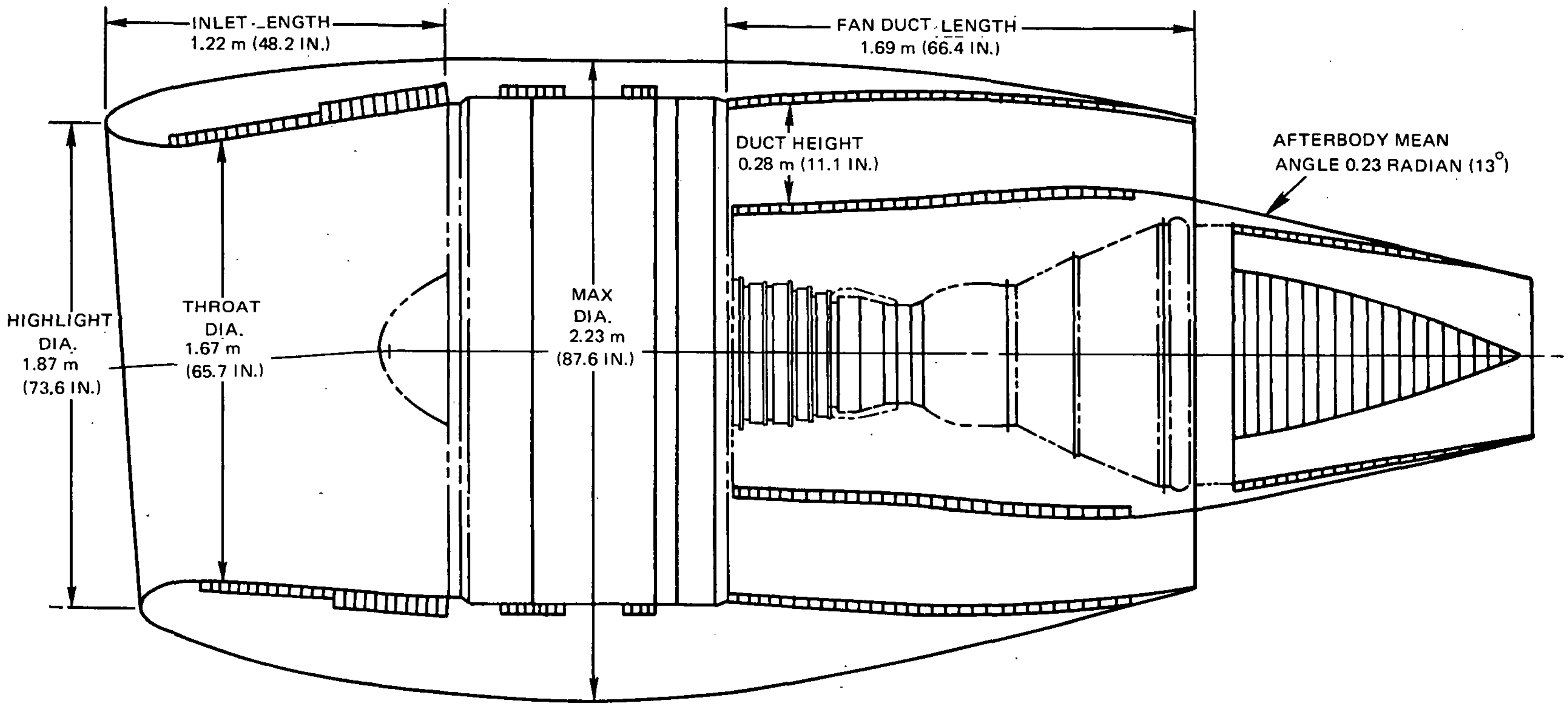



*Figure 6.3.1-1 STF 477 Engine Cross Section With High Spool Advanced Technology Concepts Identified*

TABLE 6.3.2-1

COMPARISON OF COMPONENT CHARACTERISTICS OF 1975 AND 1985 TECHNOLOGY FUEL CONSERVATIVE TURBOFAN ENGINES AT CRUISE DESIGN POINT

| | Major Changes (1975 to 1985) | Potential Benefits |
|---|---|---|
| Cruise Design Cycle Parameters | 1) Increase cycle pressure ratio from 25:1 to 45:1<br>2) Increase fan pressure ratio from 1.6:1 to 1.7:1<br>3) Increase bypass ratio from 6:1 to 8:1<br>4) Increase maximum combustor exit temperature by 111°C (200°F) | |
| Fan | 1) Eliminate part span shrouds<br>2) Improve airfoil shapes<br>3) Reduce endwall losses<br>4) 61 m/sec (200 ft/sec) higher tip speed | a) +1.8 percentage points efficiency |
| Compressor | 1) Increase pressure ratio per stage by 7 percent<br>2) Increase inlet corrected tip speed by 152 m/sec (500 ft/sec)<br>3) Improve blading<br>4) Reduce tip clearance | a) +3.3 percentage points polytropic efficiency |
| Diffuser/Burner | 1) Improve diffuser design<br>2) Reduce burner exit temperature profile<br>3) Reduce emissions | a) −1.0 percent pressure loss |
| Burner/Turbine Gaspath Materials | 1) Improve burner liner<br>2) Use monocrystal/eutectic airfoils<br>3) Use high temperature protective coatings<br>4) Improve turbine seals | a) Increased cycle pressure ratio capability<br>b) −3.6 percent chargeable cooling air |
| High-Pressure Turbine | 1) Reduce load factor<br>2) Increase speed<br>3) Reduce endwall losses<br>4) Reduce cooling air penalty<br>5) Reduce tip clearance | a) +2.9 percentage points efficiency |
| Low-Pressure Turbine | 1) Increase load factor<br>2) Improve aerodynamics<br>3) Reduce tip clearance | a) Reduced weight and cost<br>b) +1.1 percentage points efficiency |




**INLET GEOMETRY**

$$\frac{\text{HIGHLIGHT DIA.}}{\text{MAX. DIA.}} = 0.84$$

$$\frac{\text{HIGHLIGHT DIA.}}{\text{THROAT DIA.}} = 1.12$$

$$\frac{\text{INLET LENGTH}}{\text{MAX. DIA.}} = 0.55$$

**DUCT GEOMETRY**

$$\frac{\text{FAN DUCT LENGTH}}{\text{DUCT HEIGHT}} = 6.0$$

*Figure 6.3.3-1 STF 477 Acoustically Treated Nacelle*

## 6.4 BENEFITS OF THE STF 477

### 6.4.1 Fuel Savings and Economic Benefits

Airplane performance and economic trends were studied for the selected current and advanced technology turbofan engines at aircraft cruise Mach numbers in the range from 0.7 to 0.85. A Mach 0.8 trijet weight breakdown is given in Table 6.4.1-I for both the 1975 technology turbofan and for the 1985 technology STF 477. Similar weight data are given in Table 6.4.1-II for a Mach 0.8 quadjet. These comparisons at Mach 0.8 were typical of the trends over the Mach 0.7 to 0.85 range. The data are presented for the design range and the average stage length selected to represent typical in-service use of the aircraft. The fuel-used data in the tables represent a 16 percent reduction for the trijet and a 17 percent reduction for the quadjet.

Using the average stage length weight data from the tables and the economic ground rules in Appendix A, DOC and ROI were calculated for each aircraft powered by the 1975 and 1985 technology turbofans (1985 data calculated assuming a FAR 36 minus 10 EPNdB noise level). For the domestic trijet, the study showed a decrease (1975 to 1985) in DOC from 0.9314 ¢/seat-km (1.499 ¢/seat-statute mi.) to 0.9196 ¢/seat-km (1.480 ¢/seat-statute mi.), for a decrease of 1.3 percent. For the international quadjet, with the higher price fuel, DOC decreased from 0.9309 ¢/seat-km (1.4981 ¢/seat-statute mi.) to 0.8855 ¢/seat-km (1.425 ¢/seat-statute mi.), for a decrease of 4.9 percent. Similarly, ROI was found to increase by 0.32 percent for the trijet and by 1.36 percent for the quadjet. The results reflect improvements due only to propulsion advances.

The above comparison is based on a 10 EPNdB different engine noise goal for the two engines. The ability of the current turbofan engine to achieve a FAR 36 minus 10 EPNdB noise goal with current acoustic technology is seriously questioned. At best, extensive acoustic treatment, involving multiple inlet rings and duct splitters as well as wall treatment, would be required. Resultant penalties in fuel consumption and economics would be unacceptably high. On the other hand, if an STF 477 noise goal of FAR 36 were used, domestic trijet fuel usage could be reduced by an additional 1.6 percent, DOC reduced by an additional 1.0 percent, and ROI increased by 0.4 percentage points, assuming advanced acoustic technology. At equal noise, therefore, a nominal potential fuel saving of over 17 percent is estimated with advanced propulsion technology.

The advanced technology turbofan would appear to offer the potential for significant fuel conservation with small improvements in economics when based on the results of this comparison. An explanation for these trends, as affected by the propulsion variables, is shown in Figure 6.4.1-1. In this figure the propulsion variables, based on the STF 477 advanced engine characteristics, were 12.5 percent better installed cruise TSFC, 22 percent reduced engine weight, 8 percent higher engine price and 26 percent increased maintenance costs. The large reductions in fuel consumption that are possible with advanced technology are principally a result of the improvements in installed TSFC. Engine weight has only a small effect on fuel use for the systems studied. The purchase of fuel and oil represents 20 percent and 30 percent of the DOC for the trijet and longer range quadjet, respectively. This is reflected in the change in sensitivity of the DOC of the two systems shown in

TABLE 6.4.1-I

**WEIGHT COMPARISON OF CURRENT AND ADVANCED TECHNOLOGY TURBOFAN IN DOMESTIC TRIJET AIRCRAFT AT MACH 0.8**

| | 1975 Technology[1] Design Range[3] | | 1975 Technology[1] Average Stage Length[4] | | STF 477[2] Design Range[3] | | STF 477[2] Average Stage Length[4] | |
|---|---|---|---|---|---|---|---|---|
| | kg | lbm | kg | lbm | kg | lbm | kg | lbm |
| Fuel | | | | | | | | |
| Used | 25,677 | 56,607 | 5,935 | 13,085 | 21,656 | 47,742 | 4,988 | 10,996 |
| Reserve | 5,309 | 11,705 | 4,736 | 10,441 | 4,466 | 9,846 | 4,000 | 8,820 |
| TOTAL | 30,986 | 68,312 | 10,671 | 23,526 | 26,122 | 57,588 | 8,988 | 19,816 |
| Operating Empty Weight | | | | | | | | |
| Propulsion (3 Engines) | 8,313 | 18,328 | 8,313 | 18,328 | 7,046 | 15,534 | 7,046 | 15,534 |
| Structure | 30,359 | 66,929 | 30,359 | 66,929 | 29,052 | 64,049 | 29,052 | 64,049 |
| Systems | 6,834 | 15,066 | 6,834 | 15,066 | 6,589 | 14,526 | 6,589 | 14,526 |
| Furnishings and Equipment | 8,966 | 19,767 | 8,966 | 19,767 | 8,972 | 19,781 | 8,972 | 19,781 |
| Operating Items | 5,468 | 12,055 | 5,468 | 12,055 | 5,449 | 12,012 | 5,449 | 12,012 |
| TOTAL | 59,940 | 132,145 | 59,940 | 132,145 | 57,108 | 125,902 | 57,108 | 125,902 |
| Payload | 18,597 | 41,000 | 10,229 | 22,550 | 18,597 | 41,000 | 10,229 | 22,550 |
| Take-Off Gross Weight | 109,523 | 241,457 | 80,840 | 178,221 | 101,827 | 224,490 | 76,325 | 168,268 |

1. 1975 technology turbofan, FAR 36 noise level.
2. 1985 technology turbofan, FAR 36 minus 10 EPNdB noise level.
3. Design range 5,560 km (3,000 n.mi.) with 200 passengers.
4. Average stage length 1,300 km (700 n.mi.) with 110 passengers.

TABLE 6.4.1-II

WEIGHT COMPARISON OF CURRENT AND ADVANCED TECHNOLOGY TURBOFAN IN INTERNATIONAL QUADJET AIRCRAFT AT MACH 0.8

| | 1975 Technology[1] Design Range[3] | | 1975 Technology[1] Average Stage Length[4] | | STF 477[2] Design Range[3] | | STF 477[2] Average Stage Length[4] | |
|---|---|---|---|---|---|---|---|---|
| | kg | lbm | kg | lbm | kg | lbm | kg | lbm |
| Fuel | | | | | | | | |
| Used | 53,866 | 118,755 | 16,324 | 35,989 | 44,532 | 98,177 | 13,536 | 29,841 |
| Reserve | 8,279 | 18,251 | 5,305 | 11,695 | 6,668 | 14,700 | 4,164 | 9,181 |
| TOTAL | 62,145 | 137,006 | 21,629 | 47,684 | 51,200 | 112,877 | 17,700 | 39,022 |
| Operating Empty Weight | | | | | | | | |
| Propulsion (4 Engines) | 8,968 | 19,770 | 8,968 | 19,770 | 7,390 | 16,293 | 7,390 | 16,293 |
| Structure | 33,975 | 74,903 | 33,975 | 74,903 | 31,548 | 69,551 | 31,548 | 69,551 |
| Systems | 7,721 | 17,023 | 7,721 | 17,023 | 7,294 | 16,080 | 7,294 | 16,080 |
| Furnishings and Equipment | 9,056 | 19,965 | 9,056 | 19,965 | 9,066 | 19,987 | 9,066 | 19,987 |
| Operating Items | 6,299 | 13,886 | 6,299 | 13,886 | 6,249 | 13,776 | 6,249 | 13,776 |
| TOTAL | 66,019 | 145,547 | 66,019 | 145,547 | 61,547 | 135,687 | 61,547 | 135,687 |
| Payload | 19,278 | 42,500 | 10,614 | 23,400 | 19,278 | 42,500 | 10,614 | 23,400 |
| Take-Off Gross Weight | 147,442 | 325,053 | 98,262 | 216,631 | 132,025 | 291,064 | 89,861 | 198,109 |

1. 1975 technology turbofan, FAR 36 noise level.
2. 1985 technology turbofan, FAR 36 minus 10 EPNdB noise level.
3. Design range 10,200 km (5,500 n.mi.) with 200 passengers.
4. Average stage length 3,700 km (2,000 n.mi.) with 110 passengers.

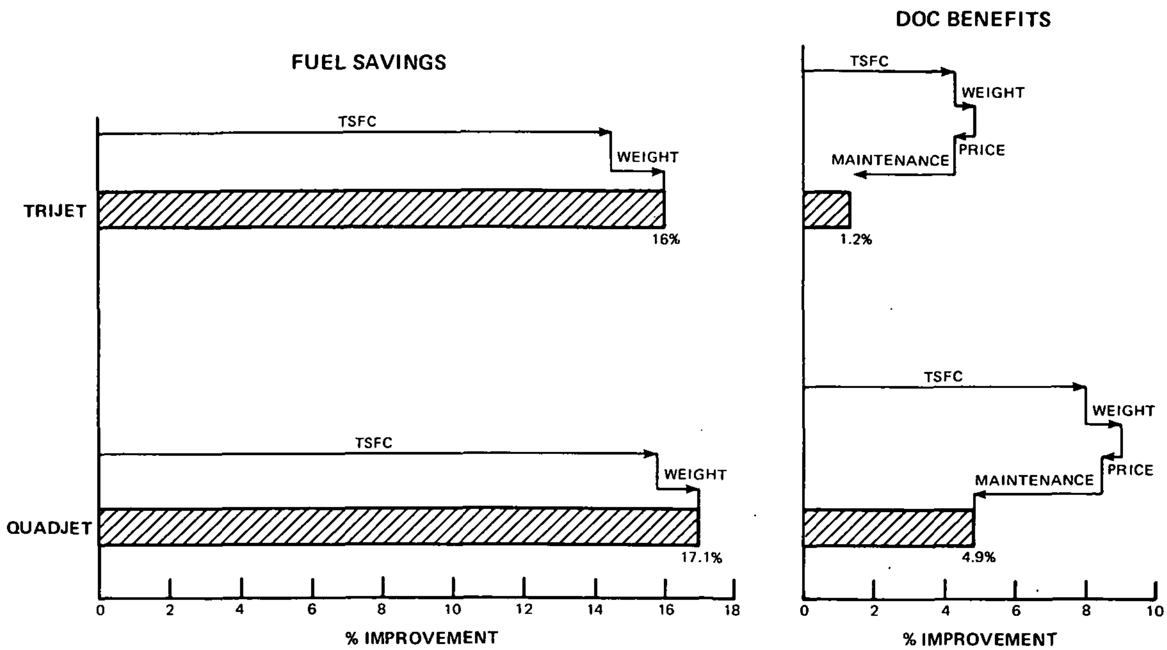



*Figure 6.4.1-1 Airplane Sensitivity to STF 477 Parameters of TSFC, Engine Weight, Engine Price, and Engine Maintenance Costs*

Figure 6.4.1-1. Engine maintenance costs (including burden) represent approximately 14 percent of the DOC for the two aircraft. Therefore, even with the large reduction in fuel purchased, an estimated 26 percent higher maintenance costs of the advanced engine nearly nullified this influence on DOC for the trijet and halves the quadjet potential benefit. When the additional effect of higher price for the advanced turbofan engine is considered, only small improvements in DOC remain, based on the baseline fuel price assumptions of 8 cents per liter (30 cents per gallon) and 12 cents per liter (45 cents per gallon) for domestic and international fuel, respectively. However, economics are extremely sensitive to fuel prices. For example, if a domestic fuel price of 16 cents per liter (60 cents per gallon) is assumed, the DOC of the advanced technology turbofan trijet system could be reduced by an additional 3 percent relative to the current technology turbofan. The results of this study serve to emphasize the need to balance the desire to conserve fuel by improving engine TSFC with the increased engine costs incurred in arriving at an economically sound engine design.

### 6.4.2 Acoustical Benefits

The definition of acoustical treatment and inlet and ducting length was based on an acoustical evaluation for the STF 477 engine. Noise estimates were made at take-off, approach, and sideline for a 116,000 kg (255,000 lbm) take-off gross weight Mn 0.8 domestic trijet and a 132,000 kg (290,000 lbm) take-off gross weight Mn 0.8 international quadjet. The engine was scaled to 78,300 N (17,600 lbf) take-off thrust required for the domestic trijet and to 68,900 N (15,500 lbf) take-off thrust for the international quadjet.

Results of the domestic trijet analysis are summarized in Figure 6.4.2-1. For the trijet, approach conditions were the most critical relative to meeting noise requirements. With present acoustical wall treatment, noise levels as low as FAR 36 minus 5.5 EPNdB are possible. For the quadjet, the take-off condition (without cutback) was the most critical in meeting the FAR 36 minus 10 EPNdB noise goal.

Technology improvements in both noise source and attenuation characteristics are required to achieve the FAR 36 minus 10 EPNdB noise level. Figure 6.4.2-1 shows the required improvements for a trijet. Fan source noise reductions of 2 to 3 EPNdB through improved airfoil design to reduce tones, broadband, and buzzsaw noise are required. Improvements in burner design are required to reduce core noise by approximately 4 EPNdB. Should these reductions not be achieved, then a core suppressor is necessary to achieve the FAR 36 minus 10 EPNdB noise level. Optimization of turbine blade and vane numbers and spacing are required to reduce turbine generated noise by 2 EPNdB take-off and 5 EPNdB approach power settings. In addition, the application of advanced acoustic treatment concepts, such as tailoring to the model structure of the noise and the use of segmented liners, is required to provide attenuation of fan and turbine noise of 2 and 3 EPNdB. For the international quadjet, the necessary technology improvements to achieve the FAR 36 minus 10 EPNdB noise goal are similar to those outlined for the trijet. These improvements could be achieved by a 1990 operational time period through intensive research and development programs in each of the noise areas.

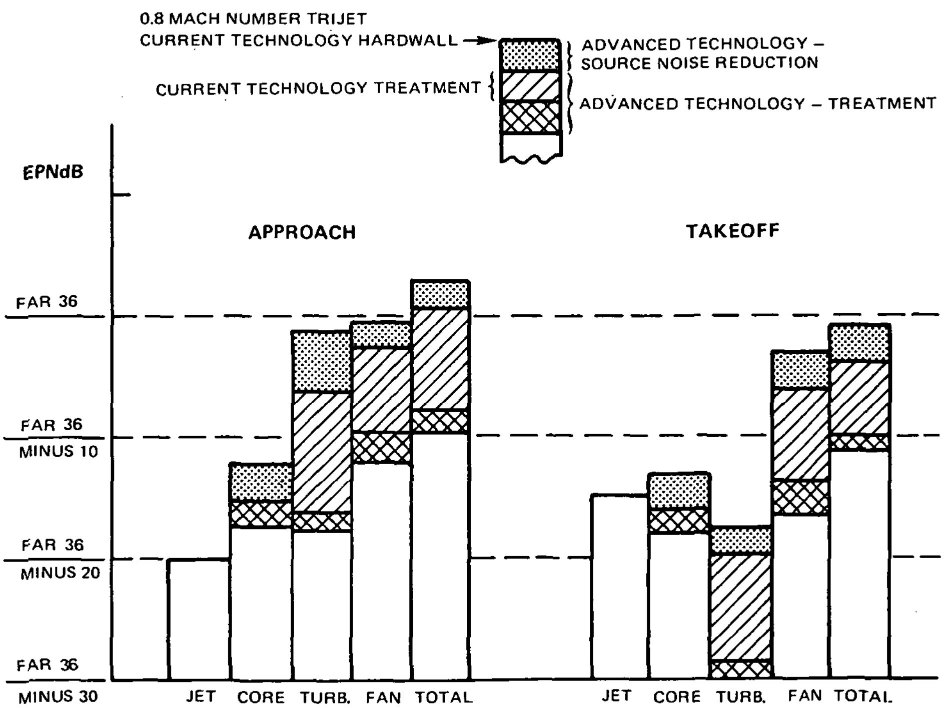



*Figure 6.4.2-1* *STF 477 Noise Generation in Domestic Trijet (Mn = 0.8) Illustrating Potential Noise Reductions with Current and Advanced Technology*

## 7.0 DISCUSSION OF RESULTS OF TASK IV – RECOMMENDED TECHNOLOGY PROGRAMS

The primary objective of Task IV was to identify technology programs required to achieve energy savings with current and advanced technology turbofan engines. This section describes the technology programs that are considered to be critical to these engines. These programs address the major engine components and material considerations and ancillary systems. Generally, the programs consist of analytical design studies, component technology substantiation, and engine demonstration. Components within an engine interact with each other, and when one component is modified, it may adversely affect another component. Modifications should therefore, be incorporated one at a time, and then in groups into an engine for demonstration testing. It is desirable to test all of the modifications simultaneously at the end of the program to assess their total affect on performance. Recommendations for interactive testing on current engines are discussed in Section 7.1.3.

The broad scope of the technology program required to bring fuel conservative current and advanced engine technologies to a state of readiness would represent an all-out effort of major proportions. A ten-year program with a cost in excess of $500,000,000 is estimated. Although the propulsion portion of the present Aircraft Energy Efficiency Program as presented by NASA is considerably less in scope than this, it will address the programs which are considered to be of highest priority and which should have a major impact on reducing the fuel consumption in current and future engines.

### 7.1 TECHNOLOGY PROGRAMS FOR REDUCING FUEL CONSUMPTION IN CURRENT TURBOFAN ENGINES

The technology requirements for developing each fuel conserving concept for current engine application are discussed in the following pages. Several additional programs in the area of operational performance retention are recommended.

#### 7.1.1 Internal Engine Improvements

7.1.1.1 High Flow Capacity Fan (JT3D)

Improvements in both fan efficiency and flow capacity have been demonstrated with the JT3D-7 fan configuration. Since the affected parts are directly interchangeable between the JT3D-3B and -7D models, there is no design effort required. However, the thrust, engine pressure ratio (EPR), exhaust gas temperature (EGT) and rotor speed relationships of the -3B at its rating points will be affected, requiring re-certification by the FAA and revision of engine operating manuals. Engine testing of this modification before re-certification would demonstrate the component and engine operational and performance changes.

7.1.1.2 Fan Blade Performance (Chamfered-Cut Leading Edge) (JT8D, JT3D)

This program would be directed toward determining the performance effects of chamfer cutting the blade leading edge to the minimum required radius. A key element in substantiating this modification is to ensure that expected performance benefits can be demonstrated with adequate blade structural integrity. The recommended program would include experimental testing of the current blade configuration to establish baseline performance levels. This would be followed by testing of several modified configurations for selection of the optimum design. The scope of the rig effort would include a determination of blade stress characteristics, leading edge shock conditions, resistance to foreign object damage (FOD), and containment characteristics. After completing the rig test phase, the selected configuration would be engine tested to ascertain the impact of the revised blade geometry on fan aerodynamic and overall engine performance.

7.1.1.3 Fan Blade Performance (Low Aspect Ratio Fan Blade) (JT9D)

A 3.8 reduced aspect ratio fan blade has been demonstrated on an experimental JT9D engine. Because of its longer chord and heavier weight blade pull stresses could result in a significant engine weight penalty. Blade aspect ratios between the 3.8 experimental blade and the 4.6 production blade would be considered to select the final configuration.

Evaluation of the selected configuration in an experimental engine program is recommended. The program would include evaluation of performance, stress, noise, and stability characteristics. Tests would be performed at sea level, in an altitude facility, and on the wing of the B52 airplane. Design modifications would be evaluated experimentally. Successful modifications would be incorporated in the engine. Parts would be procured for final acceptance testing, including FOD and containment, and engine certification.

7.1.1.4 Low-Pressure Turbine Performance (JT8D, JT3D)

Existing design methods, which were developed for the JT9D low-pressure turbine, would be applied to the JT8D and JT3D low-pressure turbine redesigns. The design approach would be to employ high efficiency turbine airfoils with controlled vortices and overlapping seals to reduce parasitic losses. Experimental rig testing would be used to verify the benefits of a controlled vortex design, airfoil stress levels, and low-pressure turbine exit profile characteristics. Engine evaluation of this modification is recommended under a combined component demonstration program in a JT8D engine to determine the interactive performance effects.

7.1.1.5 Sixth-Stage Turbine Blade Performance (JT9D)

This item is currently in production, and will be retrofitted to existing engines. It is not, therefore, considered to be part of a recommended technology program.

7.1.1.6 Turbine Exhaust Case Strut Aerodynamic Redesign (JT8D)

The JT8D turbine exhaust case struts were recambered and tested under the NASA Refan program in an effort to recover some of the performance loss resulting from residual turbine exit swirl. The general configuration of the Refan exhaust case would be used on the conventional JT8D-9 engine but with less strut recamber. Analytical studies would be required to define the desired exhaust case geometry. Engine demonstration of this modification would be required in conjunction with the other component improvements outlined in Section 7.1.3 of this report.

7.1.1.7 Resized Turbine and Primary Exhaust Nozzles (JT3D)

Rematching the JT3D engine to improve cruise performance would require flow area changes in the first and second turbine vane rows and in the tailpipe. These changes would require minor modification to existing hardware. The design and fabrication effort would therefore be minimal. The modifications must be tested at sea level and at simulated cruise conditions in the engine to evaluate the performance and stability effects.

7.1.1.8 Abradable Compressor Gaspath Seals (JT8D, JT3D)

The application of abradable tip seals in the JT3D and JT8D compressors requires evaluation of material abradability, compressor performance, and stress levels. A review of abradable seal technology is required to match the abradable materials and processes to the requirements of the various stages of the JT3D low-pressure compressor and the JT8D high-pressure compressor.

Specimens would be fabricated and selected alloy systems would be rig tested at simulated engine conditions. Abradability tests would be conducted to define wear characteristics resulting from blade-seal interaction. Erosion and oxidation tests would establish the potential endurance capability of the systems. Bond testing would establish techniques for attaching the abradable to the base material. Finally, a thermal analysis would establish the effect of the addition of the abradable on the thermal response of the seal. On the basis of the test results, a system would be selected and recommended for engine tests at each of the compressor outer seal locations.

A compressor rig evaluation program is needed to determine the aerodynamic performance and surge margin effects of the reduced tip clearances that will be possible with abradable rub strips. The effect on blade vibration stresses would be determined in the same test program.

7.1.1.9 Fan Air Cooled High-Pressure Turbine Case (JT9D)

High-pressure turbine case cooling requires nacelle cooling air re-routing that would have to be coordinated with the airframe manufacturer. This coordination effort is not expected to disclose any feasibility questions. The demonstration program can therefore proceed directly upon finalization of the design of the air distribution and impingement system similar to that used on the JT9D-59A/70 engines.

7.1.1.10 Replacement Bearing Compartment Carbon Seals (JT9D)

A wet face carbon seal system for the JT9D-7 No. 3 bearing compartment has been successfully tested in an experimental engine. JT9D engine tests of this modification as part of a combined component demonstration would provide the necessary additional technical data to verify the fuel conservative benefits.

7.1.1.11 Structural Fan Exit Guide Vanes (JT9D)

The use of steel, titanium, aluminum, and composite materials in fabricated construction would be considered for the structure of the combined fan exit guide vane/fan exit case strut of the JT9D. Static load tests of simulated exit case structures joined to an engine intermediate case are recommended to guide selection of a basic structural design approach. A case featuring the selected structural configuration would then be designed in detail. Specimens simulating each detailed structural joint would be fabricated, for example, a composite to metal transition in a fabricated case. Destructive testing of these specimens would substantiate the rigidity and durability of the proposed detailed design under static and dynamic loads. Possible case redesign would incorporate improved joint configurations. A prototype would be fabricated for static and dynamic testing in an assembly including adjacent engine cases. A structurally acceptable case should then be built into an experimental engine for durability demonstration.

A number of modifications to the pylon nose shape and vane cambers would be required to optimize pylon matching and vane performance. Each hardware modification would require testing in an altitude test facility. The selected configuration would be incorporated in the final design and parts procured for final acceptance testing.

7.1.1.12 Improved Compressor Interstage Cavity Sealing (JT8D, JT3D)

Design analytical studies are recommended to initially assess the possible approaches for reducing the cavity sizes at the inner wall between the compressor rotor and stator assemblies. Selected approaches would be rig tested for performance measurement. Specific attention would be focused on attainment of an acceptable compressor exit pressure profile, stall margin, and distortion sensitivity. Stress and interstage air properties which could be affected by the redesigned components would be measured. The final element of the program would be an engine demonstration test to assess the integrated performance of the engine with this modification.

7.1.1.13 Case-Tied Low Pressure Turbine Seals (JT9D, JT3D)

A detailed thermal response study of case-tied seals is necessary to identify the clearance pinch point and to define the minimum allowable design clearance for each seal. Analytical definition of the seal system design would then be conducted. A combination of rig testing in a simulated operating environment and full-scale engine testing would be used by hardware testing. Engine testing would be conducted as part of the JT9D component demonstration program.

**7.1.2 Installation Improvements**

7.1.2.1 Forced Mixing of Primary and Secondary Exhaust Streams (JT9D, JT8D, JT3D)

Expansion of forced mixer technology to realize the potential performance improvement available from mixing the exhaust streams of the current engines would involve a combination of analytical and experimental programs.

All current production JT3D and JT9D engines are installed in short or 3/4 length duct nacelles. To accommodate exhaust mixers, these installations would have to be modified to provide long ducts. Also, preliminary studies of structurally integrated engine/nacelles have shown that long ducts may provide advantages in reducing the deformation of engine cases due to thermal and maneuver loads. A joint P&WA-airframe manufacturer study to more accurately assess the expected energy conservation level achievable would be a necessary first step.

If a significant payoff is still indicated, an experimental static test program utilizing small scale hot flow models could be run to investigate the performance and exit velocity profile effects for potential lightweight mixer concepts. This would be followed by further external flow testing of the model to establish performance, weight, and acoustic attenuation characteristics. A complementary analytical effort will utilize data from the test programs to assist in formulation of mixer design criteria. This system will contain all of the necessary mixer trade factors including mixer and nacelle weight and cost, engine performance improvements, installed performance drag penalties, and acoustic attenuation.

Mixer weight can be minimized through extensive use of titanium. While design studies recommend the selection of titanium, candidate alloy creep and ductility properties are marginal. Therefore, alloy/process efforts are needed to develop a fully viable mill product. In addition, high creep strength sheet alloy formability and weldability studies are needed to support reduced cost mixer fabrication.

7.1.2.2 Replacement Exhaust Nozzles (JT9D, JT8D, JT3D)

The exhaust nozzle redesign concept is another exhaust system modification that indicates a potential fuel savings. This change in nozzle configuration would also necessitate nacelle aerodynamic and structural modifications, but to a much lesser extent than those required with the exhaust mixer. Even though the performance, weight, and cost implications of the nozzle redesign are expected to be relatively small, a more accurate assessment of their value should be obtained through a joint P&WA-airframe manufacturer study before this concept is pursued further. The feasibility of eliminating the JT9D primary reverser would also be evaluated.

Verification of the benefit of nozzle redesign and quantification of the actual energy conservation that can be realized would require a combined model, full scale, and flight test program. The model testing would involve running of isolated and installed small scale nozzles to evaluate their external drag and basic static internal performance. A full-scale engine test in an altitude facility would supplement the model efforts and the accuracy of the internal performance data. Final verification of the actual fuel savings would be obtained through modification of the exhaust nozzles on an aircraft and flight testing of the new configuration.

Program elements include model testing of nozzle modifications to accommodate removal of the reverser.

### 7.1.3 Component Interactive Testing (JT9D, JT8D, JT3D)

Pratt & Whitney Aircraft experience has shown that interactive effects occur within an engine when one component is modified. To assess these effects, component demonstration programs are recommended for the JT9D, JT8D, and JT3D engines.

For the JT9D engine, individual component modifications, such as the structural fan exit guide vanes and replacement bearing compartment carbon seals, would be incorporated in planned groups into a test engine, and evaluated at both sea level and altitude conditions. The test program would be conducted at both the Pratt & Whitney Aircraft Willgoos Laboratory altitude test facility and in a B-52 testbed aircraft. Special performance and stress instrumentation would be installed to monitor the operating characteristics of the modifications. Final testing would include evaluation of all the selected modifications for the JT9D engine.

The program would involve the utilization of two production engines. The first engine would be used as a baseline in a production sea level test cell and then modified and re-tested. The program would proceed to an altitude evaluation at which time the modified parts would be removed sequentially and the engine retested. The engine would then be re-assembled with special features and installed in the B-52 aircraft for flight evaluation and the parts removed and retested to assess separate component effects. Testing with this engine would include high speed x-ray testing to evaluate high-pressure turbine sealing, ingesting and containment to evaluate fan components, and a noise investigation to ensure compliance with FAR standards.

The second production test engine would also be baseline tested at sea level conditions and flight-tested in the B-52 aircraft. This approach of using two engines with the same modifications and similar test conditions would provide the necessary data base to corroborate the interactive effects of the modified components.

The component demonstration for JT8D engine component modifications would be conducted in a similar manner. This effort would be completed in two phases. The first phase would evaluate the interactive effects of the chamfered fan blades, abradable seals, and aerodynamically redesigned exhaust case struts. In the second phase of the test, the improved low-pressure turbine and interstage cavity sealing modifications would be assessed.

The recommended program for the JT3D engine would also include sea level and altitude evaluation of the suggested component modifications. Component modifications would be incorporated on a collective basis to evaluate the overall performance and stability effects.

## 7.2 TECHNOLOGY PROGRAMS FOR REDUCING FUEL CONSUMPTION IN FUTURE TURBOFAN ENGINES

The achievement of the potential fuel savings indicated in this report will require significant advancements over a wide range of technologies. The critical technologies and estimates of fuel savings in medium or long range transports relative to 1975 technology engines are listed in Table 7.2-I. The benefits shown on the table include the cycle improvements available with the technology advancement. For example, the high temperature materials benefits in the high spool will include a significant increase in cycle pressure ratio and attendent increase in cooling air temperature levels allowable with the improved materials.

**TABLE 7.2-I**

**1985 TURBOFAN TECHNOLOGY REQUIREMENTS AND POTENTIAL BENEFITS**

| | Fuel Savings Relative to 1975 Technology |
|---|---|
| • Advanced High Spool | 11% |
| High Temperature Burner-Liner and Turbine Airfoil Materials and Coatings | |
| Efficient, High Speed High Spool System | |
| • Improved Passive and Active Clearance Control Seals | 3% |
| • Advanced Low Spool | 3% |
| High Efficiency Fan<br>High Load Factor Turbine | |
| • High Strength to Density Ratio Material | 1% |
| Composites<br>Titanium Base Alloys<br>Nickel Disk Alloy | |

The technologies shown in Table 7.2-I are all engine improvements. In addition to these, achievement of significant energy savings will be possible with other technology advances listed in Table 7.2-II. Programs for these technology improvements are also outlined in this section.

**TABLE 7.2-II**

**OTHER 1985 TURBOFAN TECHNOLOGY REQUIREMENTS**

- Acoustical advances
- Operational performance retention
- Full authority, digital electronic control
- Maintenance cost reduction

### 7.2.1 High Temperature Materials and Coatings for Combustor and Turbine Airfoils

The potential for reduced fuel consumption with high pressure ratio is predicated on the achievement of higher hot section materials temperature capability to avoid large cooling air penalties. The improvements shown in the evaluation reflect an increase in burner liner and turbine blade metal temperatures of 83°C to 111°C (150°F to 200°F). An advanced oxide dispersion strengthened burner liner material and a directionally solidified eutectic blade alloy show promise for providing this potential. Complementary oxidation-erosion resistant and/or insulative coatings will also be needed for the blades, vane platforms, and outer air seals.

#### 7.2.1.1 High Temperature Burner Liner Materials

In order to provide a 45:1 cycle pressure ratio without an increase in liner cooling air, an approximately 83°C to 111°C (150°F to 200°F) higher metal temperature capability than present day conventional materials can provide will be required. Advanced alloys have the potential for the higher creep strength and higher metal temperature operating levels required beyond the best conventional liner materials available now. Although the identification of the properties of these materials is in the preliminary stage, data are available which permit projections in metal temperature capability of up to 306°C (550°F) beyond present day levels (Hastelloy X alloy levels). The recommended initial program would identify the desired composition and processing techniques required to produce an alloy which would operate, uncoated, at the desired temperature conditions. This would be followed by the development of fabrication techniques and the collection of design data leading to the fabrication and test of experimental burners. Engine tests on the most promising concepts in high temperature, high pressure environments would verify the applicability of this liner in the advanced technology turbofan burner.

#### 7.2.1.2 Monocrystal/Eutectic Materials and Improved Coatings for Turbine Airfoils

Monocrystal/eutectic alloys and improved turbine airfoil coatings show promise for higher strength, higher temperature capability for turbine blades relative to the best directionally solidified superalloys and coatings currently available. For example, a Ni-23.1 Cb-4.4 Al eutectic alloy $(\delta' + \delta)$ combined with a suitable high temperature coating would have the potential for a 50 percent increase in blade design stress or a 56°C to 111°C (100°F to 200°F) increase in metal temperature. Applied to high pressure turbine blades, this material system could lead to an increase of 20 percent in rotational speed, resulting in reduced diameter turbine designs and fewer compressor stages. In addition, turbine cooling requirements could be decreased.

The recommended programs would determine the feasibility of applying monocrystal/-eutectic alloy systems to commercial advanced subsonic turbofan engines. Alloys that would receive special attention are those with properties of low density, a high melting point, and good oxidation resistance. Creep rupture strength and ductility would be determined as a function of solidification rates and alloy composition. Candidate coating materials for the external and internal surfaces would be screened by dynamic oxidation-erosion and ductility rig testing and by inter-diffusion analysis. The most promising alloy and coating systems would be tested for general compatibility when exposed to advanced subsonic turbofan turbine environment and stress/cycle conditions. Additional considerations will include the ability to strip and recoat the various concepts.

#### 7.2.1.3 Ceramic Thermal Barrier Coatings

The development of ceramic thermal barrier coatings would permit significant reductions in turbine cooling air to improve the turbine component and cycle performance. The 278°C (500°F) or greater temperature capability of ceramics over metal alloy counterparts allows them to be exposed to the hot gas as a thermal barrier. The low conductivity of ceramics provides an insulative effect on metal backing for minimizing the cooling air requirement. However, uncertainty exists in defining and evaluating the thermal fatigue and impact resistance properties of these brittle materials.

The recommended program consists of cyclic endurance testing of ceramic coated vane platforms, turbine outer air seals, and blades in an engine environment. The research program to develop ceramic coated vane platforms and outer air seals, currently in process, would be continued. Ceramic coatings on blades have been conceptually verified by test. Additional testing is necessary to demonstrate the strength, corrosion and foreign object damage resistance, thermal fatigue properties, and repair restoration capabilities of these coatings.

### 7.2.2 Efficient High Speed High Spool System

The combination of technological advances in the compressor, burner, and high pressure turbine aerodynamics has shown significant potential for reducing fuel consumption in future new turbofans. Increased rotor speed is fundamental to this improvement. This, together with high pressure levels, imposes severe operating requirements on the main bearings and seals. Research and technology programs are required in each of these important areas if the potential improvements are to be realized.

As an intermediate step between individual burner and turbine component technology development and a high spool engine technology demonstration, the use of the NASA high pressure burner and turbine facility is possible. This would permit the concurrent development of low emissions combustors including the interactions with the downstream turbine. The ultimate testing of these components in a high spool engine is seen as a critical part of an overall program to substantiate the fuel consumption improvements of the high compression ratio system in a simulated real engine environment. Inlet pressurization and heating would be used to achieve the 45:1 pressure ratio and required temperature levels. As shown in Figure 6.3.1-1, the high spool would consist of a multistage compressor, low emissions burner, and high work, advanced technology turbine. The high spool would also be used as a vehicle for demonstrating new materials, advanced cooling techniques, active clearance control, and high speed bearings and seals.

#### 7.2.2.1 Advanced Compressor Technology

The studies described in this report have shown the desirability of higher rotor speeds. The need still exists to determine the desired aerodynamics including diffusion factor, aspect ratio, through-flow Mach number, axial velocity ratio, blading solidity, reaction, and airfoil and endwall losses.

The design optimization study and statistical parameter optimization have been initiated for an advanced multistage compressor under the NASA Advanced Multistage Axial Compressor Program. This program, as a typical example, would guide the identification of the mix of design parameters for the compressors.

Additional analytical and test programs are recommended to reduce airfoil and endwall loss levels for maximum compressor efficiency attainment. The compressor design resulting from the above programs would be tested both as an individual component and as a part of a high spool engine.

#### 7.2.2.2 Advanced Combustor Technology

The major needs for technological advances in the combustor identified in this program are for reduced emissions in conjunction with high temperature, high pressure operation. Potential improvements in inlet diffuser aerodynamics were also identified as being beneficial in reducing the fuel consumption of the engine.

In the emissions area, oxides of nitrogen are, at present, the greatest challenge in comparison with the Environmental Protection Agency 1979 or 1981 rules. This situation is aggravated by the high pressure ratio of the advanced fuel conservative engine. The recommended program consists of two elements. First, screening of the low emission burner concepts would be conducted to partially define the final arrangement. The final arrangement would also be based on expanded thermal and structural evaluations of improved film cooled structures and other advanced high effectiveness cooling techniques. Second, the burner concept would be designed, fabricated, and tested in a rig and in the advanced technology high spool engine.

Diffuser improvement programs including the evaluation of compressor-diffuser-combustor compatibility and aerodynamic optimization are recommended with the objective of developing a design approach of diffusers in the actual flow field.

#### 7.2.2.3 Advanced High Pressure Turbine Technology

Increased turbine efficiency means increased rotor speed and reduced load factor and hence increased running stress levels, rim cavity flow, and disk windage losses. Therefore, the recommended program addresses these potential limitations while improving turbine efficiency.

With the high stress levels at the blade attachments in an advanced turbine design, methods of reducing blade weight would be pursued. One approach includes the tapering of the blade chord (reducing it toward the tip) to reduce the centrifugal load. The resulting reduced tip solidity could result in a reduction in turbine performance. Solidity limits would be defined by analyses and confirmed by test.

High speed turbines also require high blade inlet flow angles with an unknown effect on the end losses. Analyses indicate that high wheel speed turbines are most susceptible to the effects of cavity purging flows than conventional turbines. This parasitic loss would be investigated experimentally. Rotating disk windage losses due to disk/air scrubbing for this turbine would also be determined empirically.

Component performance would be specified experimentally by running the turbine component in a high spool engine simulating both the gas path and non-gas path engine temperature and pressure conditions.

7.2.2.4 Advanced Bearings and Seals Technology

The higher rotor speeds coupled with the increased pressure levels of the advanced turbofan require significant advances in the main engine bearings and bearing compartment seals. The higher rotor speeds could result in bearing DN levels (bearing bore diameter times speed – mm X rpm) approaching 3 million or greater, as shown in Figure 7.2.2.4-1, in combination with main seal face speeds of 180 m/sec (600 ft/sec) or greater, as shown in Figure 7.2.2.4-2. Since, with current practice, the internal bearing compartment operating pressure is near ambient, high pressure differentials as high as $2.41 \times 10^6 N/m^2$ (350 $lbf/in^2$) could be required across the seals. Since these levels considerably exceed those of present designs, technology programs are recommended in both of these areas. First, new bearing design concepts would be tested to achieve the high speed levels with commercial life. Second, the high seal speed and pressure differential requirement would be approached by two routes. Gas film seal technology would be developed for future engine application in place of either carbon rubbing contact seals or labyrinth seals currently in use. Second, pressurization of the main bearing compartments would be evaluated and tested as a means of reducing the internal to external pressure differential.

### 7.2.3 Improved Passive and Active Clearance Control Seals

In addition to direct aerodynamic improvements to the engine components, careful control of running clearances and undesired leakage of air can lead to efficiency improvements. A 3 percent, or greater, potential fuel savings is estimated with improved sealing technology in the compressor and turbines. Effective sealing throughout the engine gas path and especially at the blade tips is crucial to the potential improvement.

Rapid engine power transients produce a differential thermal growth between rotor assemblies and cases, engine structural deflections from case temperature gradients, and aircraft flight and ground induced loads. All of these factors contribute to excessive running clearances in these critical seal regions. The program would include the definition of technology and systems to passively and actively modulate turbine and compressor blade tip

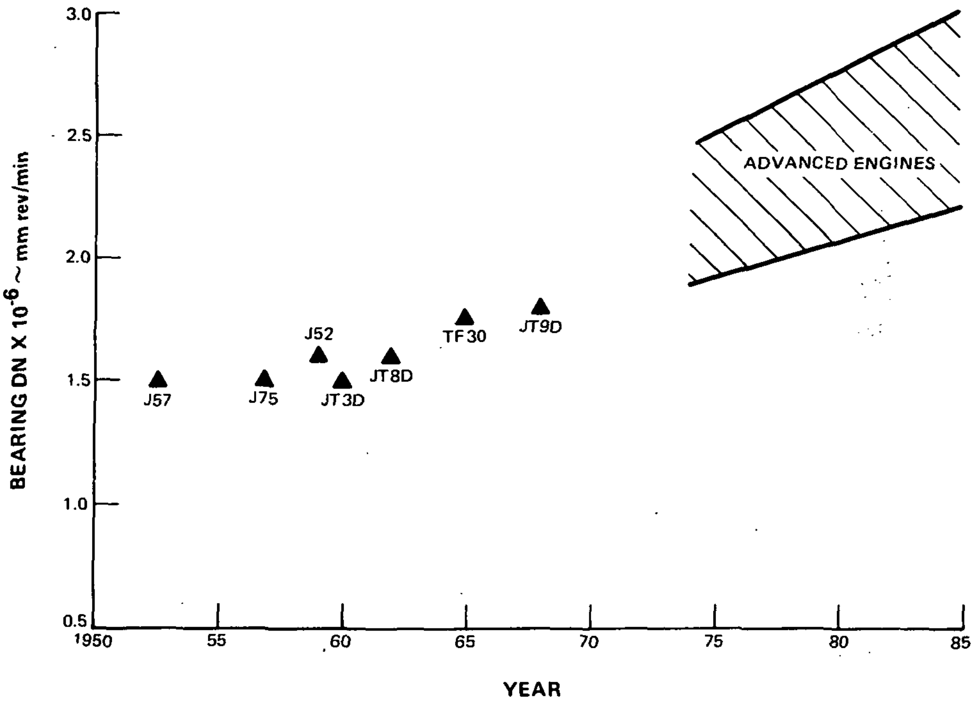



*Figure 7.2.2.4-1* *Bearing DN Levels of Current Engines and Projection to Advanced Engines*

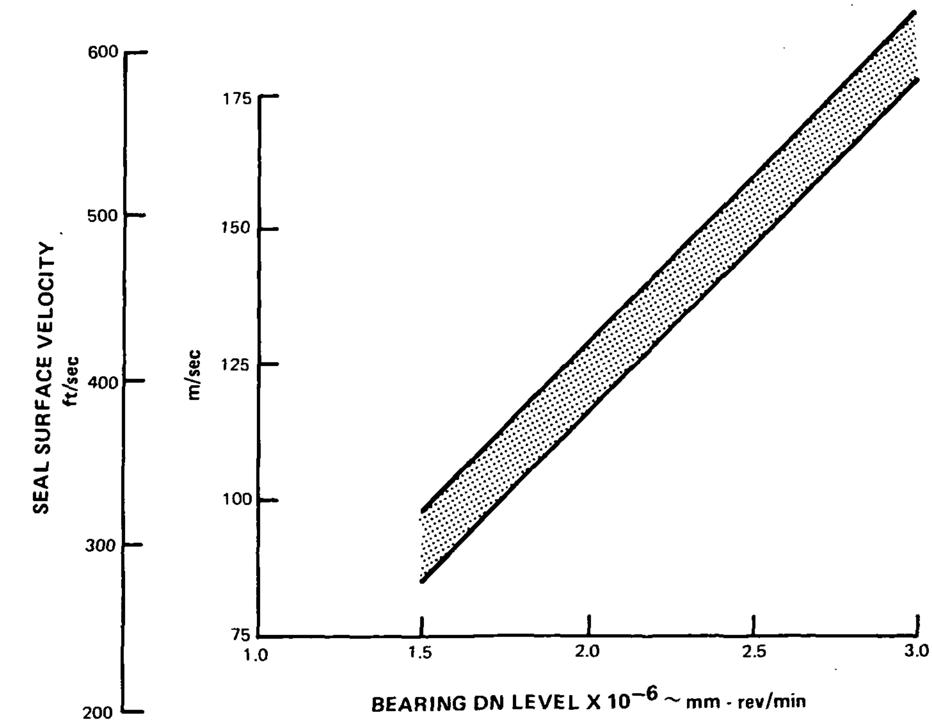



*Figure 7.2.2.4-2* *Seal Face Surface Speed Projections*

clearances throughout the flight envelope. Compressor and turbine operational characteristics that affect gas path sealing would be analyzed and various concepts would be studied to compensate for factors that contribute to operating clearance. For example, compressor blade tips may show a cruise operating clearance of 0.51 mm (0.020 in.) when designed for minimum clearance at sea level take-off conditions. The turbine is also extremely sensitive to running blade tip clearance as shown in Figure 7.2.3-1.

In order to allow tighter running clearances without incurring the risk of rub-caused catastrophic failures, the need for nondeteriorating abradable seals is becoming increasingly important. These include metallic and ceramic materials. Equally important is the development of a compatible abrasive material for use with the abradable materials. First-generation abradable sintered graded zirconia ceramic-to-metal turbine static shrouds would be tested with an $Al_2O_3$/CoNiCrAlY matrix abrasive tip in a production engine turbine. The recommended program is to continue development of these tip seal systems directed toward reduced running clearances with good hot corrosion/erosion resistance. The research and development of a ceramic as a spray-on seal should also be expanded.

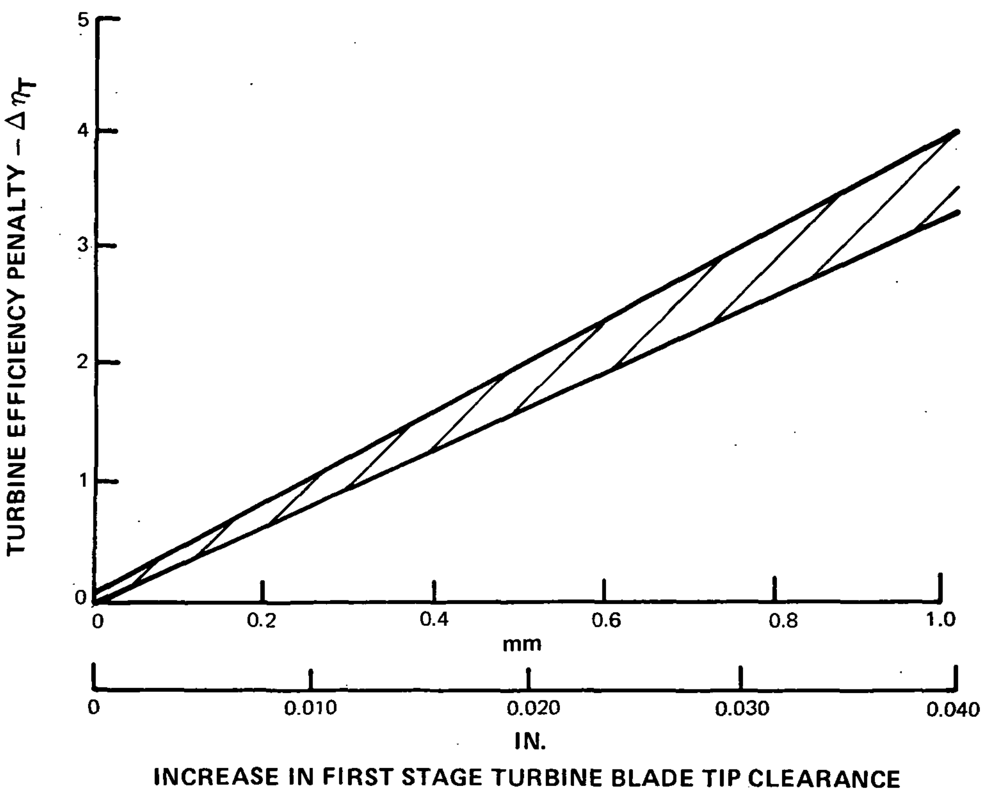



*Figure 7.2.3-1 Turbine Efficiency Penalty As a Function of Blade Tip Clearance*

In addition to passive clearance control, actively controlled tip clearance technology programs are recommended. Mechanical, pneumatic, and thermal schemes for activating tip seal controls would be appraised with a goal of reducing clearance to near zero at the cruise point. Alternate concepts for reducing the clearances would be evaluated. Cost, weight, and complexity would be considered with respect to potential fuel savings in future turbofan engines.

In addition to improved tip clearance control, reduction of parasitic leakage rates offer fuel savings. A program to improve the compressor discharge seal and to develop case flange structural and sealing design technology is recommended. The goal of this program is obtaining near zero leakage with a stable structure.

### 7.2.4 Advanced Low Spool

Technological advances in the fan/low pressure turbine rotor system could provide fuel savings of approximately three percent, relative to current engines, based on evaluations discussed in this report. A major portion of this potential improvement is contingent upon aerodynamic advances in the fan and low pressure turbine.

The use of a speed reduction gear on this spool could also offer an additional fuel consumption improvement. On the basis of the preliminary configurational comparison made in this evaluation, the geared and direct drive systems were estimated to be competitive. More detailed study is required to determine fully the relative merits of the two systems.

#### 7.2.4.1 Advanced Fan Technology

The principal requirements in fan technology advances include the reduction of airfoil and endwall losses without degrading aeroelastic integrity. Substantial potential performance benefits by elimination of part span shrouds have been defined. Flutter prediction systems to evaluate the aeroelastic effects of shroud elimination are presently in development. The benefits of advanced airfoil shapes including controlled shock blading and supercritical stators are also currently being evaluated or tested.

The recommended fan program includes expansion of these current evaluations as well as the initiation of two additional programs. These would include: (1) an evaluation of the loss reduction with improved part span shroud design and (2) study of the reduction of, or possible elimination of, part span shrouds using composite materials in the blades. Tip shroud development for possible fan rotor performance improvement is needed. The effects of blading aspect ratio selection require further evaluation. Continued evaluation of the potential of supercritical and controlled shock blading is required leading to demonstration in a fan stage. Both analytical and experimental work for endwall loss reduction are needed. Also, further prediction systems need to be refined analytically and experimentally. Composite material structural advancement is also required for fan application (see Section 7.2.5, High Strength-to-Density Ratio Materials).

7.2.4.2 Advanced Low Pressure Turbine Technology

The high bypass ratio level of the advanced subsonic turbofan will require a high load factor low-pressure turbine to minimize the number of stages. In order to provide high efficiency, the aerodynamic losses of the turbine must be minimized.

Results of the NASA 4-1/2 stage turbine program (contract NAS3-19402) will significantly increase the technology for high load factor turbines. Complementary to this program, the analytical verification and test of laminar flow airfoils to reduce airfoil losses are included as part of the recommended program.

7.2.4.3 Speed Reduction Gear Technology

The recommended gear program would address the requirements of low weight and high efficiency with commercial maintainability and reliability. Detailed design studies of a single stage gear system and selected components are required. Design and testing of a gear rig and heat rejection system would lead to full scale testing in the engine environment to verify performance and commercial acceptability.

### 7.2.5 High Strength-To-Density Ratio Materials

The utility of high strength to density ratio materials in advanced turbofan engines lies in the fuel savings resulting from reduction in propulsion system weight. Recommended programs for the advancement of lightweight materials fall into two categories, low temperature and high temperature capability. Low temperature composite materials can be considered in the fan and front end of the compressor where temperature levels do not exceed 316°C (600°F). Advanced titanium and nickel base alloys can be considered for compressor and turbine disks as well as blading for reduced weight.

7.2.5.1 Advanced Composite Materials

A low density composite material such as carbon epoxy, used in advanced turbofan fan blades, stators, and exit case, has the potential for reducing engine weight by as much as 7 percent relative to the standard titanium/aluminum construction. Two important problems with composites are foreign object damage (FOD) resistance and load transfer through joints. A program concentrating in these two problem areas is recommended. This program would consist of continual evaluation of stress distribution and FOD in lightweight composites for both metallic and non-metallic candidates. In the blades, fiber failure mode research would be expanded leading to testing of a simulated blade structure. Then testing of full scale composite airfoils and static structure would be conducted. At the conclusion of this program, the basic feasibility of a composite material for application to advanced turbofan engines will be established.

7.2.5.2 Advanced Titanium Base Alloys

High temperature titanium alloys represent a lightweight alternative to steel and nickel base alloys in compressor and low pressure turbine blades and disks. Results of this evaluation indicate a potential 3 percent reduction in engine weight through the use of advanced titanium alloys. Titanium aluminide materials have been subjected to preliminary subscale screening and processing development. The recommended program would continue the material screening culminating in a composition selection. Blades and disks are to be designed, fabricated, and tested in the appropriate component both as an individual component and in engine tests.

7.2.5.3 Advanced High Strength Nickel Disk Alloy

The use of advanced nickel alloy high pressure turbine disks are estimated to reduce engine weight by 5 percent in an advanced turbofan relative to current disk materials. A program is recommended to determine the feasibility of various approaches to meet the strength requirement for the advanced turbine disks. Some of the approaches that would be explored in this program are: new alloys to extend the high temperature creep strength capability beyond some of the research alloys that are currently being evaluated such as NASA IIB-11; composite disks including fiber-wound, multi-alloy or laminated configurations; and cast disks fabricated by hot isostatic press techniques, possibly improved by thermal-mechanical treatment such as explosive shocking. The goal for these advanced disk materials is to obtain a 20 to 30 percent greater creep strength with no compromise to either low cycle fatigue or oxidation resistance characteristics relative to current disk materials.

**7.2.6 Advanced Acoustical Technology**

Achievement of significant energy savings and compliance with stringent noise requirements such as FAR 36 minus 10 EPNdB will require advances in noise technology. Improvements in fan, combustor, turbine, and acoustic lining technology may offer the capability of achieving FAR 36 minus 10 EPNdB noise levels with peripheral treatment only, thus avoiding the weight and performance penalties associated with inlet rings or duct splitters. To develop the necessary technology, the noise programs outlined below would be conducted.

7.2.6.1 Fan Source Noise Advancements

An analytical program is recommended to develop an improved understanding of fan detailed aerodynamic design on discrete tone, broadband, and buzz-saw noise generation. This will lead to the design of fans with inherently lower source noise levels. Test programs would be required in support of the analytical study to define airfoil wake characteristics and surface pressure fluctuations of selected designs and to evaluate the analytical models. Noise prediction systems evolving from verified analytical models would form the basis for optimizing the design of a fan for both low energy consumption and low noise.

7.2.6.2 Combustion Source Noise Advancements

Two potential combustion noise sources exist – direct and indirect. The direct noise is generated by the combustor burning process. The indirect noise is generated by the convection of hot spots in the combustor exit flow through the turbine. The characteristics of these noise sources is not known for unconventional burners. Therefore, the recommended program is directed toward developing analytical models of these noise sources as well as testing burners the geometry of which is consistent with emissions requirements. Using the data from this test program, design procedures would be developed to optimize the fuel conservation engine burners for minimum noise (and emissions).

7.2.6.3 Turbine Source Noise Advancements

Turbines with high stage loadings will most likely be required. The noise characteristics of such turbines are not well defined. The program for turbine source noise reduction should include the development of a noise prediction system for high work output turbines. The prediction system should include rotor/stator interaction effects and account for flow swirl and temperature changes as well as the noise attenuation across the turbine stages. Cascade tests are recommended on high stage loading blades to define the wake characteristics. Based on the results of these programs, design and testing of a single stage turbine rig to verify the predictions may be required.

7.2.6.4 Noise Attenuation Advancements

In addition to noise source reductions, improved attenuation of fan, core, and turbine source noise is required. A program directed toward increasing the attenuation in each of these areas at least 2 EPNdB without increasing the treated area is recommended.

The objectives of the program are to develop more effective inlet and fan duct liner designs, develop low frequency core noise absorption capability, improve attenuation through the turbines, and increase the effectiveness of tailpipe liners in the attenuation of turbine noise. The program would consist of several elements: (1) Develop an analytical model for turbine attenuation of low frequency sound including experimental verification using a turbine rig and low frequency sound source; (2) Develop a prediction system capable of defining the fan model structure and techniques to measure modal structure in order to provide input to modal models that can tailor liner design to the modal structure – as part of the fan test program, a liner would be fabricated and tested to evaluate design procedures; (3) Develop low frequency muffler concepts with greater degrees of freedom to tailor liners to specific attenuation spectra. Selected liners would be designed and tested in a reverberation chamber for attenuation verification.

### 7.2.7 Full Authority, Electronic Digital Control

Generally, there are no changes that can be made to hydromechanical turbine engine controls to improve engine fuel consumption of conventional fixed geometry turbine engines. Improvements to be gained through the use of electronic controls in general for turbine engines will primarily be reduced pilot work load, through automatic rating control, as well as improved hot section life by blade temperature limiting and over-shoot protection.

The advent of full authority digital electronic propulsion controls, however, presents possibilities for significant fuel savings when coupled with aircraft control systems. Such a coupled system could include several features:

- Wing-to-Wing Thrust Balance
- Mach Number Hold
- Altitude Hold
- Minimum Trim Drag
- Automatic Climb Control
- Automatic Rating Control

Airline studies show that these features can provide up to several percentage points reduction in fuel usage. For example, flying 1.22 km (4000 ft) below optimum altitude can burn up to 0.6 percent additional fuel. Similarly, an error of 0.01 Mach number from nominal can result in as much as a 0.7 percent increase in fuel consumption. Trim drag, due to trim tab deflection, can contribute 0.5 to 1 percent to increased fuel burned. Obviously, these small percentages result in substantial amounts of fuel in a year's time. Therefore, a concerted effort should be undertaken to explore the fuel conservation benefits that might be derived from an overall aircraft/propulsion control integration concept.

Pratt & Whitney Aircraft is presently conducting extensive research and development activity in the area of digital electronic controls applicable to the fuel conservative turbofan. An additional study program is recommended. This program would include the definition and evaluation of the benefits of an integrated airplane/engine control system using digital electronic turbine engine controls and digital aircraft controls. It would be logical to include both airframe and engine manufacturers in this study. Multivariable and adaptive control logic techniques would be used for the coupled control design study, permitting automatic minimizing of fuel consumed during any portion of the flight. This study program could be expanded to include demonstration testing in a suitably modified aircraft using an existing P&WA digital control system to demonstrate some of the fuel saving benefits.

### 7.2.8 Reduced Maintenance Costs

The impact of designs to improve specific fuel consumption have a tendency to increase engine price and engine maintenance cost, reducing the potential cost benefits of low TSFC. Efforts to reduce maintenance cost are suggested for inclusion in overall NASA plans and specific program objectives in order to ensure that future technology efforts produce meaningful and practical results.

The cost of replacement and repair of high pressure turbine parts consumes the bulk of airline maintenance dollars. Improvements in this area would be part of this study. An additional study program directed at defining the cost of turbine maintenance is therefore recommended. In addition, conceptual design studies of lower cost turbine airfoil designs or longer life, repairable airfoil designs are also recommended.

## 7.3 PERFORMANCE RETENTION PROGRAMS FOR CURRENT AND ADVANCED ENGINES

### 7.3.1 Current Engines

#### 7.3.1.1 Load Sharing (JT9D)

Preliminary studies have suggested that structural load sharing integration of the engine and nacelle can reduce engine case and shaft deflections, which are suspected of being the major cause of short term performance deterioration in modern high bypass ratio engines. A joint P&WA-airframe manufacturer design study is recommended to determine the extent of integration that is feasible with the JT9D engine and its existing installations, and to estimate the performance improvement possible. The x-ray investigation of a running JT9D engine under simulated flight loading, described in Section 7.3.1.2 below, would provide essential information. The use of long fan ducts, exhaust mixer, and composite materials would also be considered in conjunction with the study of structural load sharing.

Testing of the modified structural configuration would include: static load tests of individual components to determine deflections, ultimate strength, and failure mode; x-ray tests of the running engine with simulated loads to determine clearances; and ground operation of the engine-nacelle package to evaluate structural and thermal compatibility.

#### 7.3.1.2 Diagnostic Engine Testing (JT9D, JT8D, JT3D)

Short term deterioration, which occurs during the first few hours of operation of an engine on an airplane, is believed to be the result of deflection of engine cases and rotors due to thermal or maneuver loads encountered in flight. The deterioration appears to be more severe in the modern high bypass ratio, high pressure ratio engines, which might be explained by the tighter clearances built into them and their increased sensitivity to leakage. A program to verify the cause of short term deterioration of the JT9D, using also information from JT3D and JT8D testing, is recommended. Special instrumentation would be installed in several operational JT9D engines/nacelles to measure the loads and temperatures that the engines encounter in airline service. X-ray investigation of the blade and seal clearances occurring in JT3D, JT8D and JT9D engines while they are running under simulated flight loads and temperatures would be conducted in an existing P&WA x-ray facility. The test results would be used in the JT9D Load Sharing Nacelle effort described in Section 7.3.1.1 above.

#### 7.3.1.3 Long Term Deterioration (JT9D, JT8D, JT3D)

The recently completed study by American Airlines on the deterioration of JT3D and JT8D engine performance with time determined that a substantial amount of the performance losses occurring over the long term could be associated with the deterioration of compressor and fan performance (ref. 5). A similar study effort is recommended for the JT9D engine. One of the recommendations of the study was a back-to-back series of engine tests to determine the extent of the performance losses that could be recovered by either replacement and/or refurbishment of compressor and fan airfoils and the extent to which the recovery could be achieved on a cost effective basis. Alternatively, a series of compressor cascade and rig tests could be undertaken to define the effects of the various mechanical condition differences observed in compressor airfoils of the JT8D, JT3D and JT9D engines. This approach would require additional time to complete but would be more beneficial in providing guidance in the design of advanced compressors for the engines of the future. The results would also help define the relationship between mechanical condition and performance losses in current compressors, and hence guide repair development and/or airfoil scrap criteria based on aerodynamic performance.

Steel and titanium materials have been used in compressor blades and stators over the past two decades. Steel airfoils have been coated with either cadmium, nickel cadmium or aluminum for oxidation/corrosion protection. Titanium has been used in the uncoated condition. Reduction in performance deterioration over the thousands of hours of engine operation has shown the need for increased resistance to erosion of the airfoils by particles in the air. A technology base for airfoil coatings systems having this capability is partially developed and should be pursued further now that the need for reduced fuel consumption is vital to gas turbine engine operation. Some metallic and non-metallic type coating systems have been evaluated on a limited basis in laboratory scale tests and further work is required to develop these systems further and expose them to more realistic rig and engine environments.

### 7.3.2 Advanced Engines

Fuel conservative, high bypass ratio engines possess a noticeably wasp-waisted cross section, a characteristic which makes them highly sensitive to deflections of engine cases under load and to the effects of losses in component performance from increased clearances after rubs. The recommended investigatory program is to study the early deterioration and performance loss in current engines in an attempt to isolate where and when the performance loss occurs. A combined program of study involving an airline, an airframe manufacturer, and an engine manufacturer should increase visibility in this area. Recommended parallel programs would address analytically the aircraft installation induced load, the effects of mounting arrangements, thermally or centrifugally induced interference, and other factors, to determine the probable losses in performance. This should be followed by tests of a fully cowled engine in an x-ray facility to validate the analytical techniques and provide the necessary guidance to the advanced fuel conservative engine program.

Based on the results of an industry study of JT3D and JT8D engines, long term performance deterioration is suspected to be mainly a result of loss in compressor flow capacity and efficiency because of erosion, corrosion, and foreign object damage, together with the effects of resultant blending repairs. The recommended program would consist of cascade and rotating rig testing based on inspection of long time parts of engines to establish the mechanical changes which are responsible for long time deterioration. A parallel program on erosion modelling to provide prediction tools should be included.

Multi-stage compressor rig performance testing should be contemplated to ensure that the impact of stage-to-stage variations is fully understood. The scope of this testing should address the stage interactive effects on the overall compressor to assure that small changes in the individual stage performance are not accompanied by large mismatching losses. In addition, the program should include evaluation of compressor blade erosion/corrosion resistant coatings and mechanical restoration techniques to remedy deterioration.

Since the solution to both short and long term deterioration could lead to a revised design approach to engine construction and installation, evaluation of alternate approaches to design is required to establish the favored approaches. Flight testing of the resultant prototype configuration to demonstrate performance retention could represent the final phase of the overall program.

# 8.0 CONCLUSIONS

This report summarizes the results of the study of Turbofan Engines Designed for Low Energy Consumption. Near-term technology for reduced fuel consumption in current turbofan engines was considered for the JT9D-7, JT8D-9 and JT3D-3B engines that power many subsonic transport aircraft now in service. Advanced technology, projected for 1985, was considered for reducing fuel consumption in future turbofan engines for subsonic transports of the 1990's. This section presents the conclusions that were drawn from the results of this study program.

**Reducing Fuel Consumption in Current Turbofan Engines**

- Fuel consumption reductions of 6.5, 3.3 and 7.3 percent may be achieved for the JT9D-7, JT8D-9 and JT3D-3B, respectively, using near-term technology. These reductions may be achieved through internal engine and installation modifications. The modifications will have to be substantiated through rig and engine testing.

**Reducing Fuel Consumption in Future Turbofan Engines**

- Application of advanced technology (1985) to turbofan engines designed for use in the 1990's has the potential to reduce fuel consumption by 15 percent relative to turbofans currently in service.

- To realize the potential fuel consumption benefits projected in this study, aggressive research and evaluation programs are required to develop critical advanced technology areas.

- Emissions, noise levels, engine cost and ease of maintainance must be considered in the development of advanced technology for future turbofan engines. This is required to make these engines acceptable from an environmental and overall economic standpoint.

# APPENDIX A

## AIRCRAFT CHARACTERISTICS AND CALCULATIONS USED IN ADVANCED TECHNOLOGY TURBOFAN EVALUATION

This appendix presents the airplane aerodynamics, weight, and pricing calculations, including the engine nacelle, used to evaluate the advanced technology turbofan engines. Also included are the economic groundrules used to evaluate both the current and advanced engines.

### Aircraft Aerodynamics for New Engine Evaluation

Profile drag predictions were made by the component buildup method, in which the drag coefficient $C_D$ is:

$$C_D = C_{DP\,min} + \Delta C_{DP} + C_{DI} + \Delta C_{DWD}$$

where $C_{DP\,min}$ is the minimum profile drag coefficient, $\Delta C_{DP}$ is the incremental variation of profile drag coefficient due to lift, $C_{DI}$ is the ideal induced drag coefficient, and $\Delta C_{DWD}$ is the subsonic wave drag coefficient. The ideal induced drag coefficient was computed by the standard formula for an elliptically loaded wing, $C_{DI} = C_L{}^2/\pi\overline{AR}$, where $C_L$ is the lift coefficient and $\overline{AR}$ is the wing aspect ratio. Figure A-1 illustrates schematically this drag buildup procedure. Drag coefficients are referenced to wing planform area.

Skin-friction drag coefficients, based on the Prandtl-Schlichting equation for turbulent boundary layer over a flat plate, were computed for the wing, tail, and fuselage. These coefficients were modified by the effects of wing and tail thickness ratios (thickness to chord, t/c), fineness ratio, and compressibility effects to estimate $C_{DP\,min}$. A typical profile drag variation with flight speed is shown in Figure A-2. The additional variation in profile drag ($\Delta C_{DP}$) with changes in the lift coefficient is based on correlations with wing sweep, thickness, and camber. The subsonic wave drag coefficient $C_{DWD}$ is a function of the flight Mach number relative to the critical Mach number and lift coefficient. The high speed drag characteristics are shown quantitatively in Figure A-3.

Trends of critical Mach number assumed with quarter chord wing sweep angle ($\Lambda_{C/4}$) and thickness ratio of the supercritical airfoils are shown in Figure A-4. The level of supercritical technology used was consistent with that used for the Advanced Technology Transport (ATT) studies under NASA contract NAS3-15550. The drag rise characteristics assumed for these wings are shown in Figure A-5 as a function of lift coefficient and Mach number relative to critical Mach number.

Wing geometry trends are depicted in Figures A-6, A-7, and A-8. Wing designs were selected on the basis of minimizing fuel consumption. Results of Pratt & Whitney Aircraft studies have indicated that for minimization of typical mission fuel, cruise Mach number should be 0.06 to 0.04 below the wing critical Mach number. Therefore, for any cruise Mach number, a quarter-chord wing sweep and thickness ratio combination could be determined (with use of Figure A-4) based on this criterion.

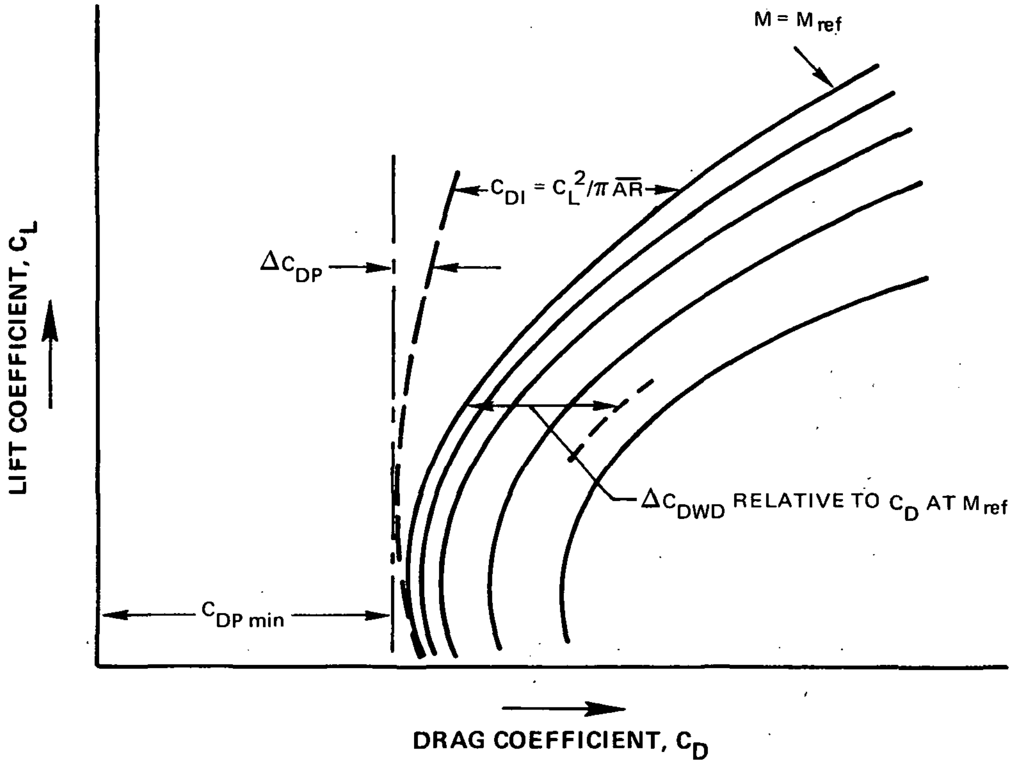



*Figure A-1* *Drag Polar Construction Procedure*

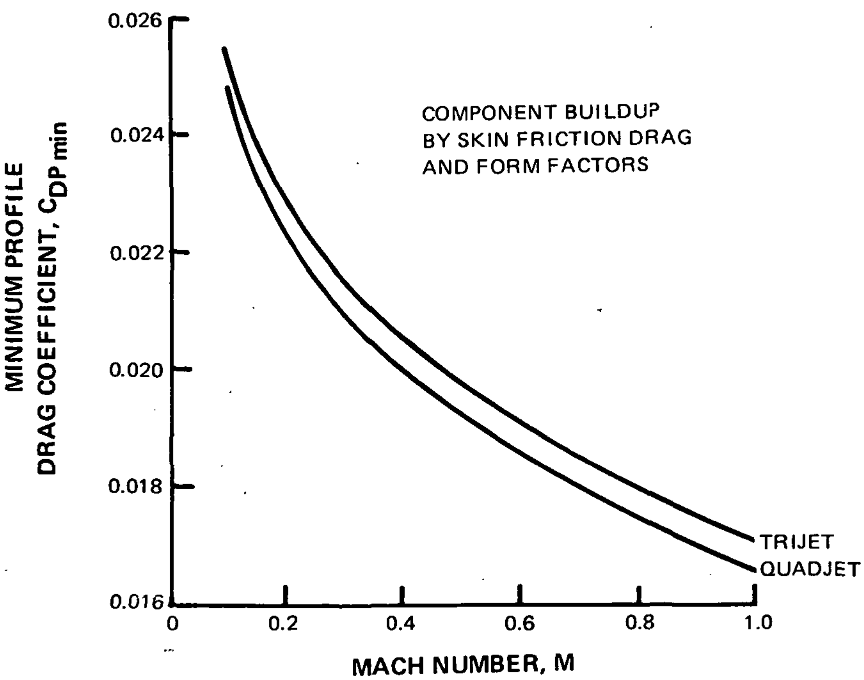



*Figure A-2* *Typical Minimum Profile Drag Coefficient for Altitude of 9,144 m (30,000 ft)*

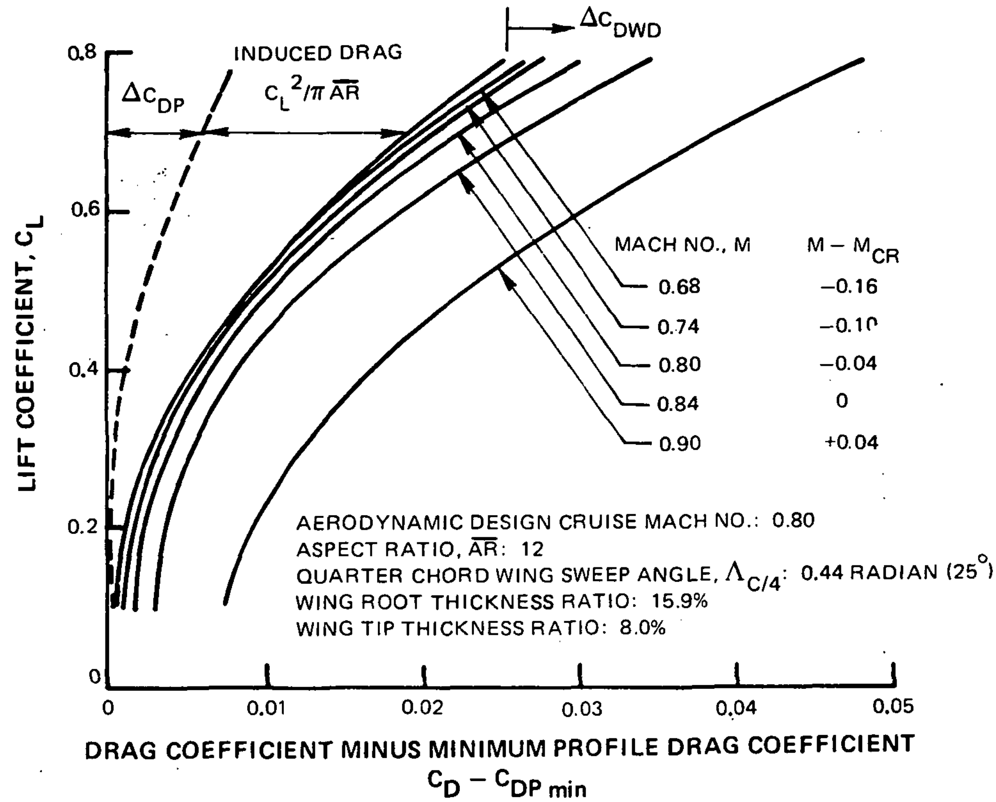



*Figure A-3 High Speed Drag Characteristics*

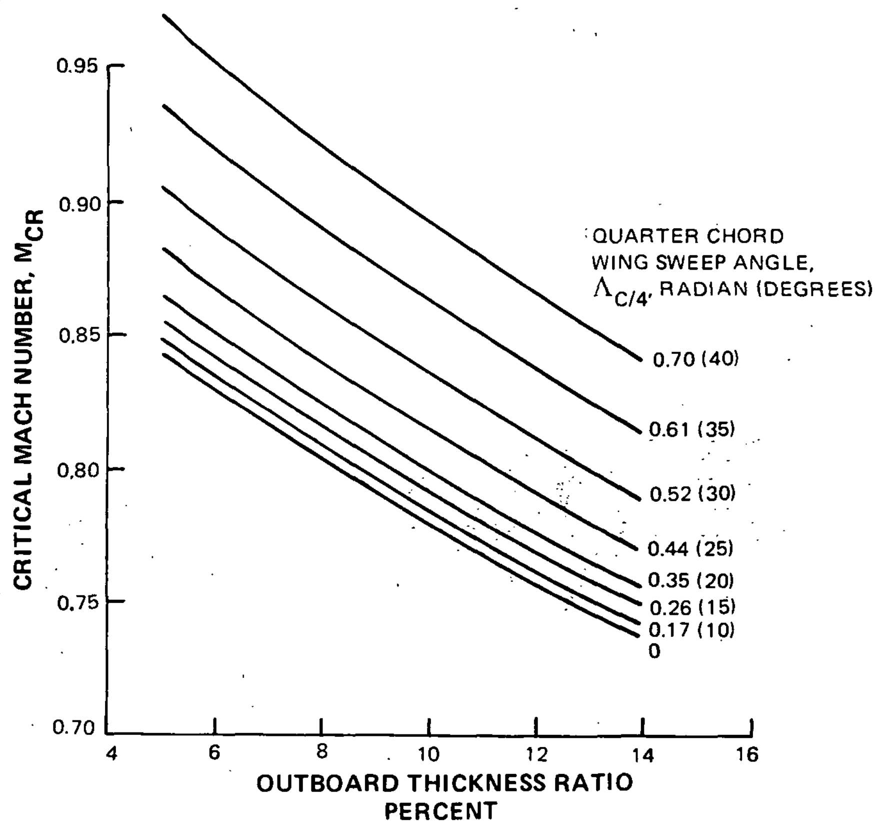



*Figure A-4 Supercritical Airfoil Technology, Lift Coefficient $C_L = 0.40$*

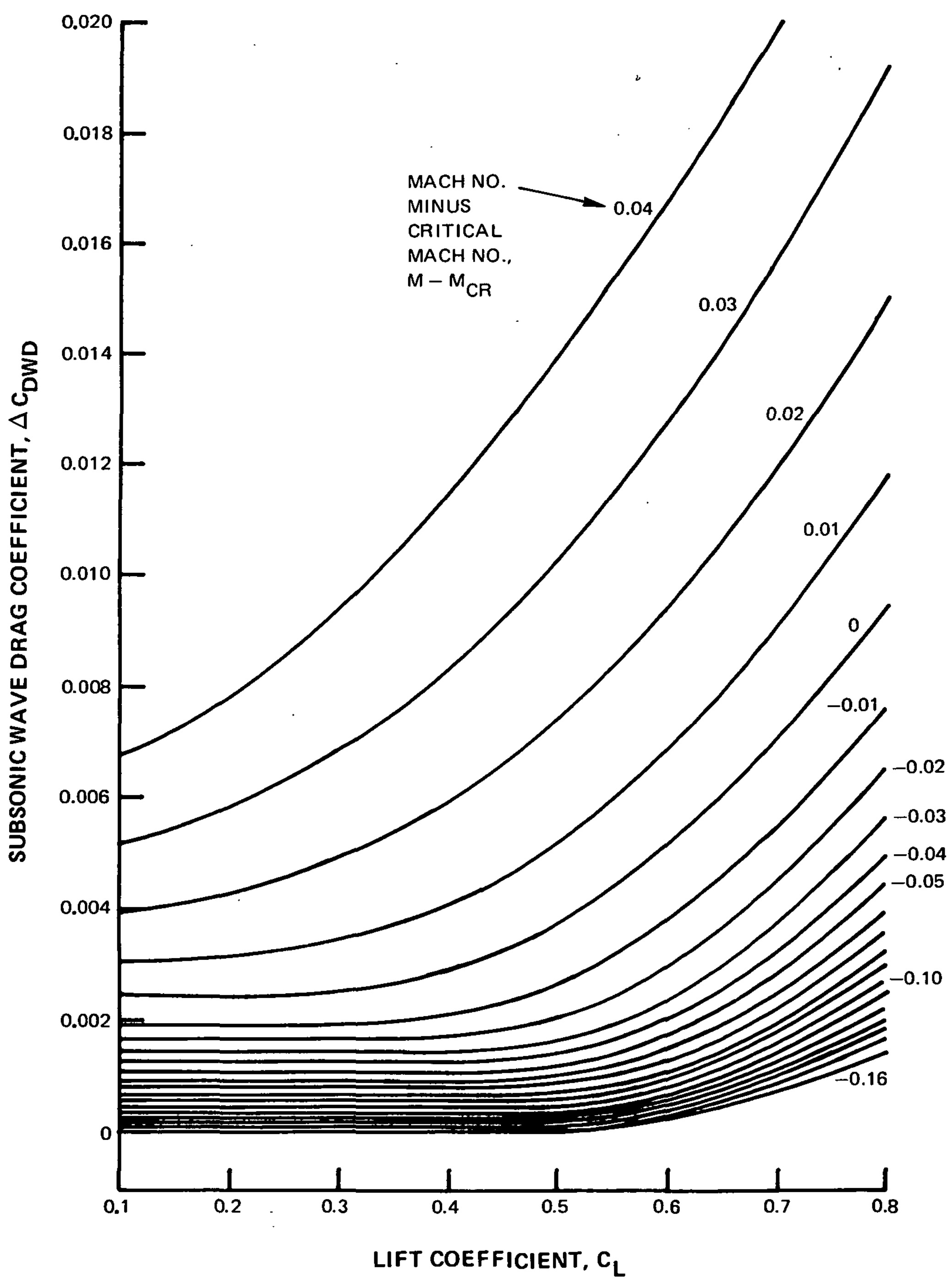



*Figure A-5 Drag Rise Characteristics of Wings*

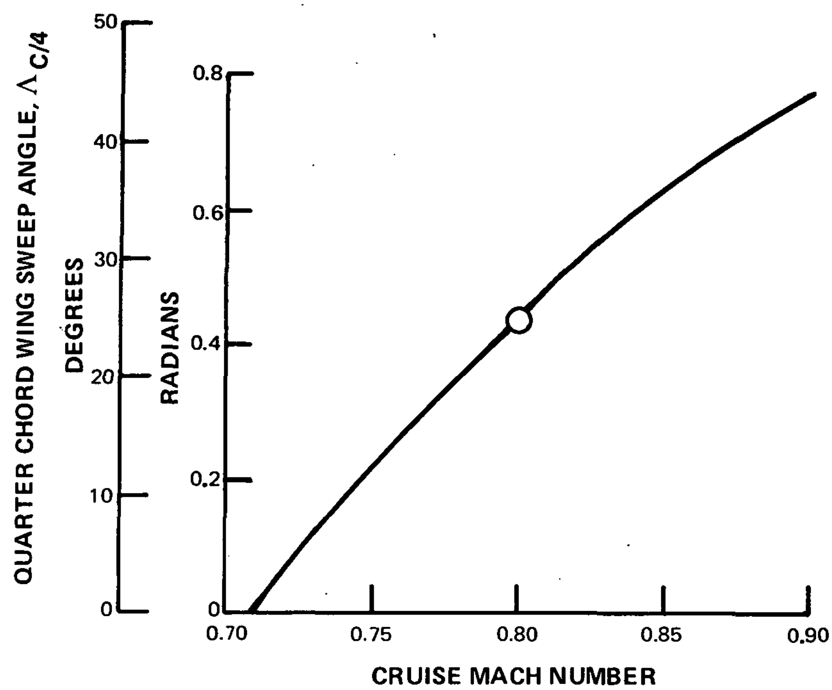



*Figure A-6* *Wing Quarter Chord Sweep Trends*

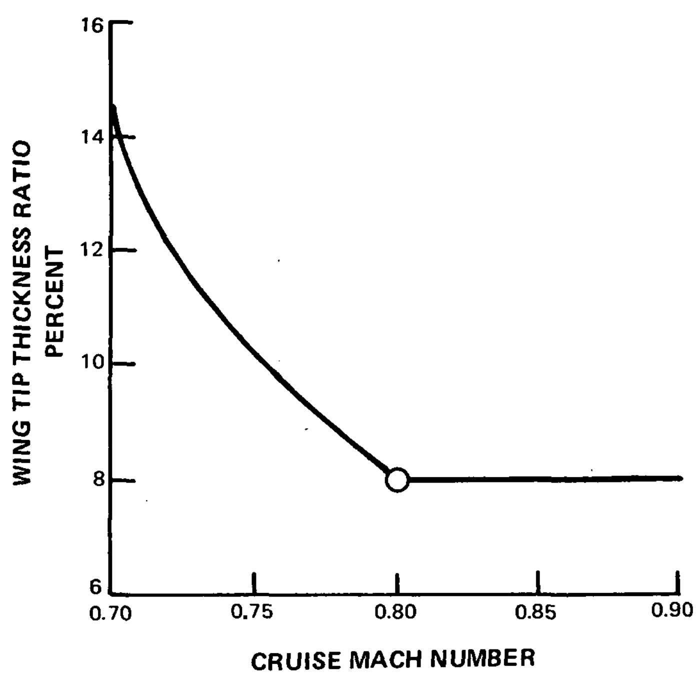



*Figure A-7* *Wing Thickness Ratio Trends*

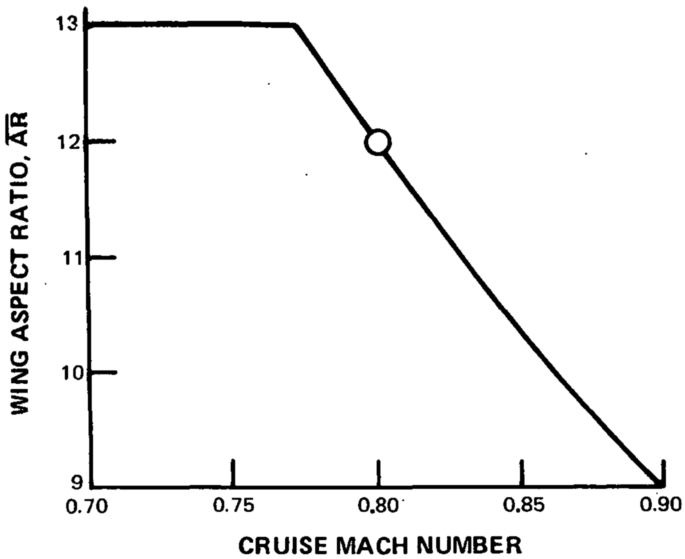



*Figure A-8 Wing Aspect Ratio Trends*

Wing aspect ratio and weight were based on an equation which related these variables with the wing parameters of sweep and thickness ratio as follows:

$$\text{Wing Weight} = K_1 \left( \frac{\overline{AR}^{0.8}}{(t/c)^{0.4} \cos \Lambda_{C/4}} \right)^{K_2}$$

The term in parentheses is called the wing weight parameter, and $K_1$ and $K_2$ are empirical constants. The fact that this parameter is approximately the same for all current aircraft led to the method for determining aspect ratio of advanced aircraft. A wing weight parameter of 2.9, reflecting the high aspect ratio wing, was used to determine aspect ratio up to a limiting value of 13.0 for the study aircraft (ref. 6).

Available industry information indicated a practical lower limit to wing outboard thickness ratio of 8 percent. This lower limit was assumed at Mach numbers of 0.8 and higher. Wing loadings were selected for the study aircraft to minimize fuel consumption within takeoff distance and approach speed limits. The wing geometries, listed in Table A-I, were selected based on these analyses.

**Aircraft Weights for New Engine Evaluation**

A component buildup method was used to estimate aircraft weight. Correlations of aircraft component weights, as related to component geometric and physical characteristics, were used for predicting the weight of all of the aircraft structural items and systems (electronic, aircraft, and fuel).

## TABLE A-I

## AIRCRAFT CHARACTERISTICS FOR SELECTED DESIGN CRUISE SPEEDS

| | Mach 0.75 | Mach 0.80 | Mach 0.85 |
|---|---|---|---|
| **Wing Characteristics** | | | |
| Takeoff Wing Loading, Trijet, $N/m^2$ ($lbf/ft^2$) | 5308 (121.2) | 5583 (116.6) | 5027 (105) |
| Takeoff Wing Loading, Quadjet, $N/m^2$ ($lbf/ft^2$) | 6368 (133) | 6607 (138) | 6464 (135) |
| Quarter Chord Sweep, radian (degrees) | 0.22 (12.5) | 0.44 (25) | 0.63 (36) |
| Aspect Ratio | 13.0 | 12 | 10.4 |
| Taper Ratio | 0.33 | 0.33 | 0.33 |
| Root Thickness Ratio, % | 18.2 | 15.9 | 15.9 |
| Tip Thickness Ratio, % | 10.3 | 8 | 8 |
| **Horizontal Tail Characteristics** | | | |
| Quarter Chord Sweep, radian (degrees) | 0.305 (17.5) | 0.52 (30) | 0.72 (41) |
| Aspect Ratio | 4.65 | 4.03 | 3.2 |
| Taper Ratio | 0.35 | 0.35 | 0.35 |
| Average Thickness Ratio, % | 10.6 | 9.5 | 7.9 |
| Ratio of Horizontal Tail Area to Wing Area, Trijet | 0.18 | 0.175 | 0.172 |
| Ratio of Horizontal Tail Area to Wing Area, Quadjet | 0.229 | 0.246 | 0.225 |
| **Vertical Tail Characteristics** | | | |
| Quarter Chord Sweep, radian (degrees) | 0.305 (17.5) | 0.52 (30) | 0.72 (41) |
| Aspect Ratio | 1.2 | 1.0 | 0.76 |
| Taper Ratio, Trijet | 0.7 | 0.7 | 0.7 |
| Taper Ratio, Quadjet | 0.35 | 0.35 | 0.35 |
| Average Thickness Ratio, % | 11.6 | 10.5 | 9.0 |
| Ratio of Vertical Tail Area to Wing Area, Trijet | 0.195 | 0.183 | 0.163 |
| Ratio of Vertical Tail Area to Wing Area, Quadjet | 0.182 | 0.186 | 0.163 |
| **Engine Location Characteristics** | | | |
| Percent Span, Trijet | 27 | 28.3 | 35 |
| Percent Span, Quadjet, Inboard | 33 | 34 | 37 |
| Percent Span, Quadjet, Outboard | 57 | 60 | 64 |
| **Nominal Fuselage Characteristics** | | | |
| Length, Trijet, m (ft) | 48.2 (158) | 48.2 (158) | 48.2 (158) |
| Length, Quadjet, m (ft) | 45.7 (150) | 45.7 (150) | 45.7 (150) |
| Height, m (ft) | 5.24 (17.2) | 5.24 (17.2) | 5.24 (17.2) |
| Width, m (ft) | 5.03 (16.5) | 5.03 (16.5) | 5.03 (16.5) |
| Number of Aisles | 2 | 2 | 2 |
| Seat Pitch, First Class, m (in.) | 0.97 (38) | 0.97 (38) | 0.97 (38) |
| Seat Pitch, Tourist, m (in.) | 0.86 (34) | 0.86 (34) | 0.86 (34) |
| Number of Passengers, First Class | 30 | 30 | 30 |
| Number of Passengers, Tourist | 170 | 170 | 170 |

The equations used for structural weight estimates are based on regressions of current, aluminum structure aircraft data. These equations were adjusted to predict composite structure weights. Table A-II shows the precentage reduction in weight of the airframe structural components assumed by composite substitution (ref. 7, 8).

**TABLE A-II**

**DIRECT SUBSTITUTION OF COMPOSITE STRUCTURAL COMPONENTS FOR ALUMINUM STRUCTURE**

| Component | Weight Reduction~percent |
|---|---|
| Fuselage | 15.5 |
| Tail | 12.7 |
| Wing | 24.6 |

Weights of furnishings and equipment, and operating items are primarily functions of the number of passengers, the number of crew personnel, cargo volume, fuel capacity, and range.

## Nacelle Geometry for New Engine Evaluation

Table A-III and Figure A-9 describe the geometry of the Task II and the STF 477 engine nacelles. Boattail angles, external cowl shapes, and afterbody geometry were based on the DC-10-40 nacelle. An external plug was used in the primary exhaust stream.

The STF 477 Task III nacelle design incorporated features developed in an ongoing P&WA nacelle study. The basic changes in this nacelle, relative to the nacelles of the parametric engines, were in the nacelle afterbody and plug. Extension of the afterbody length and increased fan cowl boattail angle provided sufficient closure to meet the primary stream exit area requirements without resorting to an external plug. The STF 477 inlet design was based on the considerations of low drag and low noise. The inlet contours provided a good compromise between the opposing requirements of low spillage drag, and low inlet weight and surface area. Inlet length was established to allow adequate noise suppression treatment to meet a total noise requirement of FAR 36 minus 10 EPNdB.

## Nacelle Weight for New Engine Evaluation

Nacelle weight estimates were based on correlations of data of current aircraft and engines. Cowl (inlet, fan, boattail, side, and afterbody) weights were estimated by multiplying a correlated area density ($kg/m^2$, $lbm/ft^2$) of the cowl component by its associated surface area. The weights were reduced by 10 percent for composites. Thrust reverser weights were made proportional to fan stream airflow. Engine accessories weights were made proportional to the primary stream air flow. Engine mount weights were assumed to be proportional to the bare engine weight. Wall treatment weights for noise reduction were a function of the treated areas. Pylon weights were correlated against thrust, nacelle diameter, and the distance between the engine and the wing.

TABLE A-III

NACELLE GEOMETRY

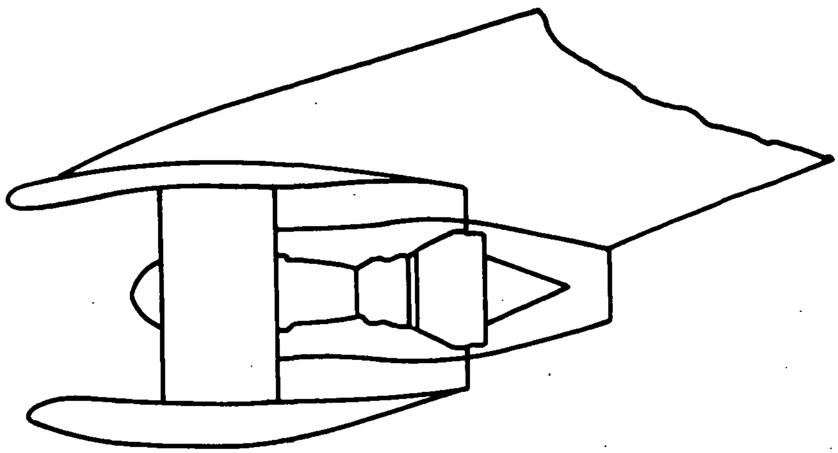
| | Task II Parametric Engines | STF 477 |
|---|---|---|
| Ratio of Highlight Diameter to Maximum Nacelle Diameter* | 0.85 | 0.84* |
| Ratio of Inlet Length to Maximum Nacelle Diameter | 0.60 | 0.55 |
| Ratio of Highlight Area to Throat Area | 1.30 | 1.25 |
| Ratio of Throat Area to Fan Face Area | 0.94 | 0.94 |
| Engine Corrected Airflow/Fan Face Area, Design Cruise, kg/sec/m$^2$ (lbm/sec/ft$^2$) | 203 (41.5) | 203 (41.5) |
| Fan Face Mach Number, Design Cruise | 0.6 | 0.6 |
| Throat Mach Number, Design Cruise | 0.67 | 0.67 |
| Ratio of Afterbody Length to Low Turbine Diameter | 0.84 | 1.03 |
| Minimum Afterbody Clearance Around Turbine, m (in.) | 0.13 (5.0) | 0.05 (2.0) |
| Minimum Nacelle Clearance Around Fan Case, m (in.) | 0.10 (4.0) | 0.10 (4.0) |
| Position of Fan Duct Exit, Ahead of Low Turbine Flange, m (in.) | 0.33 to 0.38 (13 to 15) | 0.08 (3) |
| Fan Maximum Boattail Angle, radian (degrees) | 0.175 (10.0) | 0.201 (11.5) |
| Afterbody Boattail Angle, radian (degrees) | 0.436 (25.0) | 0.454 (25.0) |
| Plug Half Angle, radians (degrees) | 1.31 (75) | – – – |

*Without provision for accessory package. Inclusion of accessory package changes value to 0.77 for STF 477 nacelle.

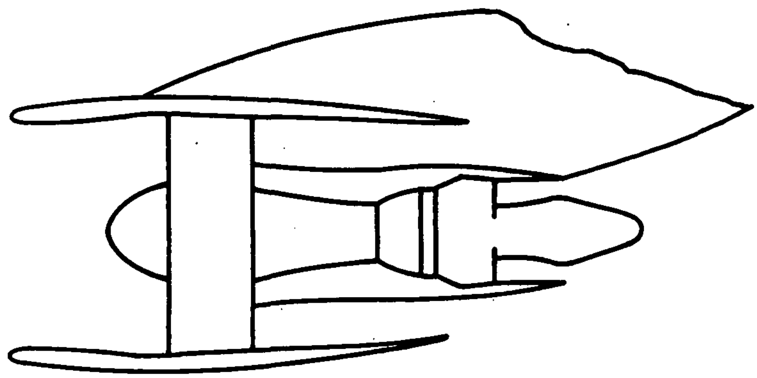
## Nacelle Drag for New Engine Evaluation

Total nacelle drag was computed by summing isolated fan cowl drag, pylon drag, and wing nacelle interference drag. Isolated cowl drag accounted for the profile plus spillage drag of the fan cowl (inlet, fan case covering, and fan cowl boattail). External drags of the afterbody cowl and the plug were accounted for in the nozzle performance data.

Pylon drag was estimated by the component build-up method for wing profile drag (described at the beginning of Appendix A). Interference drag was computed as a function of the fan exit diameter and the distance between the wing and the engine.



**STF 477 ENGINE**



**TASK II PARAMETRIC ENGINES**

*Figure A-9* *Comparison of STF 477 Nacelle Configuration With Task II Parametric Engines Nacelle Configuration (Not to Scale)*

About 45 percent of the total pod drag was fan cowl drag. Pylon drag accounted for approximately 29 percent, while interference contributed about 26 percent of the pod drag. These percentages were found to be approximately constant for all the parametric engine cycles. The fraction of total airplane drag due to the propulsion system varied from 5 to 5.5 percent for 5.0 bypass ratio engines, to 8.5 to 9 percent for the 20.0 bypass ratio engine cycles.

## Nacelle and Aircraft Pricing Method

Nacelle prices were based on regressions of current aircraft data. Engine and fan cowl prices, engine mount prices, and pylon prices were assumed to have the same cost per kilogram as the airframe, i.e., approximately $242.50/kg ($110/lbm). Thrust reversers were priced at $362.50/kg ($164.50/lbm), and accessories were priced at $319.50/kg ($145/lbm).

### Economic Groundrules

Direct operating cost (DOC) and return on investment (ROI) are used as measures of economic attractiveness. The methods used for predicting DOC are based on ATA formulae, reports of airframe and airline companies, cost estimating relationships (such as those developed by the Rand Corporation), and Pratt & Whitney Aircraft estimates of engine-related DOC components. These methods are consistent with those of NASA CR-134645 (ref. 3). Table A-IV shows the components of DOC and values of some of the factors used to compute them.

The economic model used to compute ROI required estimates of indirect operating costs (IOC), as well as DOC. Indirect operating cost calculations were based on the method described on page 271 of reference 3 and on the 1970 Lockheed method. The formulae used to calculate the various IOC components are shown in Table A-V. The method of calculating ROI is shown in Table A-VI.

## TABLE A-IV

## FACTORS USED IN CALCULATION OF DIRECT OPERATING COST

- Crew cost: Dollars per block hour are a function of take-off gross weight (TOGW) and cruise speed.
- Fuel: Block fuel per block hour times 8 ¢/liter (30 ¢/gal.), domestic, and 12 ¢/liter (45 ¢/gal.), international.
- Oil: Block fuel per block hour times 0.16 ¢/liter (0.6 ¢/gal.), domestic, and 0.24 ¢/liter (0.9 ¢/gal.), international (2% of block fuel cost).
- Insurance: 1% of flyaway price, per year.
- Airframe maintenance labor: $7.30 per manhour; manhours per block hour a function of airframe weight and average flight time.
- Airframe maintenance materials: Function of airframe weight and average flight time.
- Engine maintenance labor: $7.30 per manhour, manhours per block hour a function of average flight time and engine design.
- Engine maintenance materials: Function of engine design, size, and average flight time.
- Maintenance burden: Equal to sum of airframe and engine maintenance material and labor costs.
- Depreciation: 15 years to 0 residual value, includes 6% airframe spares and 30% engine spares. For retrofit cases (Task I) the depreciation period is 12 years for the 747 and 8 years for the 727 and 707 airframe and engines.
- Airframe price, millions of mid-1974 dollars

$$= 0.207W_A^{0.87}(Q/250)^{-0.42} + (8.6/Q)W_A^{0.89} + 0.003S + 0.600$$

- $W_A$ is the AMPR* airframe weight in kilograms divided by 453.6 (or AMPR weight in lbm divided by 1000).
- Q = quantity of airplanes = 300
- S = number of seats per airplane = 200
- ATA formula for utilization: Block hours per year = 4275 (BT + 0.3)/(BT + 1.3) + 475
- BT = block time = flight time + 0.25 hours
- Revenue load factor: 55 percent
- Typical trijet mission stage length: 1300 km (700 n. mi.)
- Typical quadjet mission stage length: 3700 km (2000 n. mi.)

*AMPR (referring to the Aeronautical Manufacturers' Planning Report) is an aircraft weight concept. Essentially, AMPR weight is the take-off gross weight less payload, engines, furnishings, fuel, instruments, electrical and other accessory equipment, and parts and fluids replaced at regular maintenance intervals. This concept is defined more completely in reference 9.

## TABLE A-V

## FACTORS USED IN CALCULATION OF INDIRECT OPERATING COST

### Mid-1974 Dollars

| | Domestic Trijet | International Quadjet |
|---|---|---|
| **Cabin Attendant, dollars per block hour** | | |
| Standard body aircraft | 20.2S/27 | 23.6S/27 |
| Wide body aircraft | 20.2S/27 + 20.2 | 23.6S/27 + 23.6 |
| **Aircraft Servicing, dollars per flight** | | |
| Fueling and cleaning | 0.78 $W_L$ | 1.91 $W_L$ |
| Landing fee | 0.43 $W_L$ | 1.05 $W_L$ |
| Aircraft control | 71 | 174 |
| **Ground Equipment and Facilities, dollars per flight** | | |
| Maintenance and burden | 0.44 $W_L$ | 0.84 $W_L$ |
| Depreciation and amortization | 0.47 $W_L$ | 0.90 $W_L$ |

**General and Administrative**

For both the domestic and international aircraft, general and administrative costs were assumed to be 6 percent of the total of DOC, cabin attendant, aircraft servicing, and ground equipment and facilities costs.

**Definition of Symbols**

S is the number of seats per aircraft (200).

$W_L$ is the maximum landing weight in kilograms divided by 453.6 (or max. landing weight in lbm divided by 1000).

## TABLE A-VI

### FACTORS USED IN CALCULATION OF RETURN ON INVESTMENT

- Basic ROI formula, mid-1974 dollars:

$$\frac{\text{ROI}}{1-(1+\text{ROI})^{-15}} = \frac{\text{Annual Cash Flow}}{\text{Initial Investment}}$$

- Annual Cash Flow = Revenue + Depreciation – DOC – IOC – Taxes
- Initial Investment, trijet = (1.06 X airplane cost) + (3.9 X engine price)
- Initial Investment, quadjet = (1.06 X airframe cost) + (5.2 X engine price)
- *Initial Investment terms: 100 percent purchase at delivery*
- Revenue: Dollars per passenger-kilometer (mile) based on Airline Operators' Guide data
- Taxes: Income and other taxes equal to 50 percent of net earnings, with no investment tax credit

## APPENDIX B

## LIST OF SYMBOLS AND ABBREVIATIONS

AMPR Aeronautical Manufacturers' Planning Report (see Table A-IV)
AR Aspect ratio
ATA Air Transport Association
ATT Advanced Technology Transport
Ave. Average
BT Block time, hours
$C_D$ Drag coefficient
$C_{DI}$ Ideal induced drag coefficient
$\Delta C_{DP}$ Incremental variation of profile drag coefficient due to lift
$C_{DP\ min}$ Minimum profile drag coefficient
$\Delta C_{DWD}$ Subsonic wave drag coefficient
$C_L$ Lift coefficient
$C_v$ Nozzle thrust coefficient, equal to gross thrust divided by ideal thrust
CO Carbon monoxide
Compr. Compressor
Dia. Diameter
DOC Direct operating cost
ΔDOC Change in direct operating cost
DN Bearing bore diameter times speed, mm-rev/min
ECCP Experimental Clean Combustor Program
EGT Exhaust gas temperature, °C (°F)
EPA Environmental Protection Agency
EPNdB Equivalent perceived noise decibels
EPR Engine pressure ratio
FAA Federal Aviation Administration
FAR 36 Federal Aviation Regulations Part 36
F-L-H Fan/low-pressure compressor/-high-pressure compressor
FOD Foreign object damage
ft Foot
gal. Gallon
Geom. Geometry
HP High pressure
hp Horsepower
in. Inch
IOC Indirect operating cost
ISA International Standard Atmosphere
$K_1$, $K_2$ Wing weight empirical constants
kg Kilogram
km Kilometer
lbm Pounds mass
lbf Pounds force
LP Low pressure
M Mach number
$M_{ref}$ Reference Mach number
$M_{CR}$ Critical Mach number
m Meter
mm Millimeter
Max. Maximum
min Minute
Mn Mach number
N Newton
n. mi. Nautical mile
$NO_X$ Oxides of nitrogen
Q Quantity of airplanes
rev Revolution
ROI Return on investment
S Number of seats per airplane
st. mi. Statute mile
t/c Thickness-to-chord ratio
THC Total hydrocarbons
TSFC Thrust specific fuel consumption
TOGW Take-off gross weight
Turb. Turbine
Vorbix Vortex burning and mixing
$W_A$ AMPR weight in kilograms divided by 453.6 (AMPR weight in lbm divided by 1000)
$W_L$ Maximum landing weight in kilograms divided by 453.6 (maximum landing weight in lbm divided by 1000)
Wgt. Weight
Δ Incremental
$\Delta\eta_T$ Turbine efficiency penalty
$\Lambda_{C/4}$ Quarter chord wing sweep angle

## REFERENCES


1. Anon.: Phase I JT3D/JT8D Refan Economics Study. The Boeing Co. report D6-41205, July 1973. Contract NAS3-16815.

2. Anon.: Variable Pitch Fan Characteristics. United Technologies Corp., Hamilton Standard Div. report QF39-950, April 1974.

3. Sallee, G.P.: Economic Effects of Propulsion System Technology on Existing and Future Transport Aircraft. NASA CR-134645, 1974. American Airlines, Inc.

4. Anon.: Control of Air Pollution From Aircraft and Aircraft Engines. Environmental Protection Agency. Federal Register, Vol. 38, No. 136, Part II, 1973.

5. Sallee, G.P., Kruckenberg, H.D., and Toomey, E.H.: Analysis of Turbofan Engine Performance Deterioration and Proposed Follow-On Tests. NASA CR-134769. American Airlines, Inc. Contract NAS3-18537.

6. Anon.: Fuel Conservation Possibilities for Terminal Area Compatible Aircraft. NASA CR-132608, March 1975. Boeing Commercial Airplane Co. Contract NAS1-12018.

7. Anon.: Advanced Transport Technology Program. General Dynamics, Convair Aerospace Div. report FZM-5779, 27 August 1971. Contract NAS1-10702.

8. Anon.: Study of the Application of Advanced Technologies to Long-Range Transport Aircraft, Interim Oral Status Report. 1 September 1971. Lockheed-Georgia Co.

9. Levenson, G. S. and Barro, S. M.: Cost Estimating Relationships for Aircraft Airframes. The Rand Corporation, report No. RM-4845-PR, May 1966.

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