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Propulsion Study for Small Transport Aircraft Technology (STAT)

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16. Abstract

The objective of the Small Transport Aircraft Technology (STAT) study was to define future research and advanced technology effort required for propulsion systems for the next generation of small, short-haul type commuter aircraft operational by 1990.

Propulsion requirements were determined for 0.5 and 0.7 Mach aircraft based on studies conducted by NASA-Ames and Lockheed California Company. Sensitivity studies were conducted on both these aircraft to determine parametrically the influence of propulsion characteristics on aircraft size and direct operating cost (DOC). Candidate technology elements and design features were identified and parametric studies conducted to select the STAT advanced engine cycle.

Trade-off studies were conducted to determine those advanced technologies and design features that would offer a reduction in DOC for operation of the STAT engines. These features were incorporated in the two STAT engines. A benefit assessment was conducted comparing the STAT engines to current technology engines of the same power and to 1985 derivatives of the current technology engines.

Research and development programs were recommended as part of an overall technology development plan to ensure that full commercial development of the STAT engines could be initiated in 1988.

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TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF TABLES

Table	Title	Page
LXVII	Alternate stage length mission results--low speed aircraft (customary units)	130
LXVIII	Technology comparison--low speed aircraft	134
LXIX	Critical technology elements for STAT research and development	144
LXX	Recommended STAT technology programs	168

SUMMARY

Detroit Diesel Allison (DDA) completed the Propulsion Study for Small Transport Aircraft Technology (STAT) for the National Aeronautics and Space Administration (NASA) Lewis Research Center (LeRC), under Contract NAS3-21995.

This study defined the future research and technology efforts that are most appropriate for propulsion systems for the next generation of small, short-haul type transport aircraft anticipated for the 1990 time frame, along with their expected benefits.

The propulsion requirements for the STAT aircraft were based on studies conducted by the NASA Ames Research Center and Lockheed California Company (LCC). The two reference aircraft were each 50-passenger, twin-engine turboprop transports designed for 1111.2 km (600 nm) range with reserves. The NASA Ames airplane cruised at 0.47 M_N with an initial altitude capability of 6278.9 m (20,600 ft). The LCC airplane cruised at 0.7 M_N with 10,973 m (36,000 ft) initial altitude capability. Typical cruise altitudes for minimum direct operating cost (DOC) for the average route segment of 185.2 km (100 nm) was 3048 m (10,000 ft) and 4572 m (15,000 ft), respectively, at design Mach number. A brief summary of additional aircraft characteristics follows:

	Low speed aircraft	High Speed aircraft
Take-off gross weight (TOGW)--kg (lbm)	17,927 (39,523)	18,291 (40,325)
Take-off distance--m (ft)	1,219 (4,000)	1,219 (4,000)
Power per engine--kW (shp)	1,767 (2,369)	3,573 (4,790)

Sensitivity studies were conducted on both of these aircraft to determine parametrically the influence of propulsion characteristics on aircraft size and cost. Using a distance of 185.2 km (100 nm) and \$0.264/L (\$1.00/gal) fuel, a 10% change in the following engine parameters produced a percentage change in aircraft DOC as indicated:

Sensitivity parameter	Δ% DOC low speed aircraft	Δ% DOC high speed aircraft
Specific fuel consumption (sfc)	3.5	4.1
Maintenance cost	1.3	1.0
Weight	0.3	0.3
Length, height (avg)	0.3	0.2

Thus, sfc and engine maintenance costs were determined to be the engine parameters that have the greatest effect on aircraft DOC.

A typical DOC breakdown is shown in Figure 1, which indicates that the portion of DOC attributable to fuel costs is 29% at $0.264/L ($1.00/gal) with engine maintenance costs representing 14%. Depreciation and insurance costs for the engine are less than 5% of the total. If fuel is assumed chargeable to the engine for the purpose of this study, the engine's share of the total DOC is 48%. If fuel goes to $0.528/L ($2.00/gal), the fuel cost becomes even more dominating and drives the engine's share to 60%. Thus, improvement in fuel economy has a significant influence on DOC.

Candidate advanced technology elements (circa 1990) and design features were identified and screened, and parametric studies were conducted to select an appropriate engine cycle. Sensitivity data were expanded to include the effect of engine design variables such as component efficiencies, pressure drop, and cooling air quantity on engine characteristics of sfc, weight, and cost. These sensitivities were used with those linking the engine characteristics to aircraft DOC to aid in the screening process. The studies were conducted at 1790 and 3579 kW (2400 and 4800 shp) for the low speed and high speed airplane, respectively. A pressure ratio of 20:1 and a turbine rotor inlet temperature (RIT) of 1506 K (2250°F) were selected for both engines for the purpose of identifying technology improvement requirements. The cycle was selected for minimum DOC for the 185.2 km (100 nm) route segment.

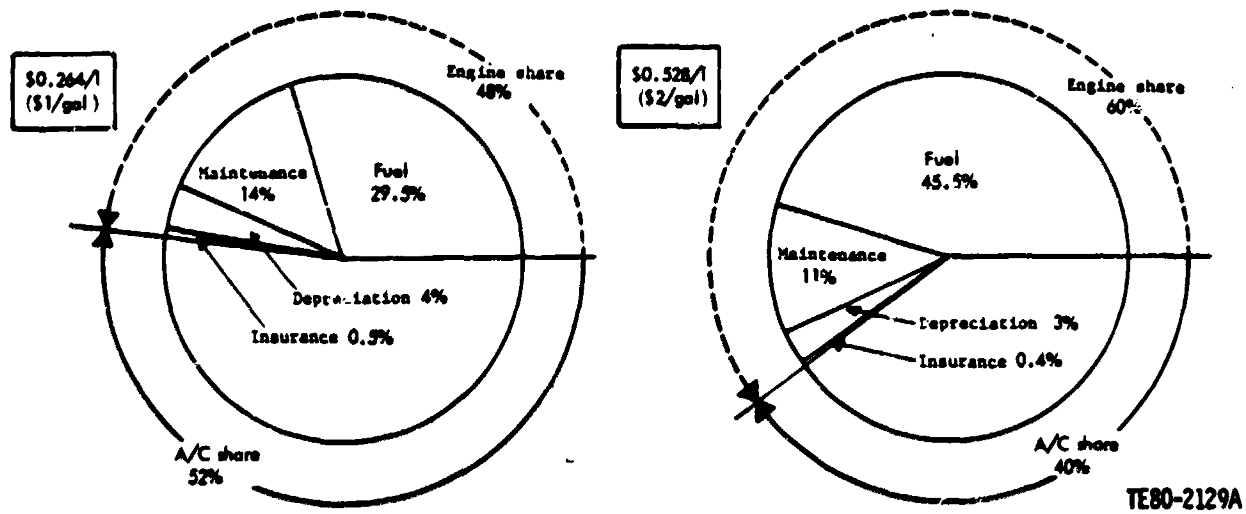


Figure 1. Typical DOC breakdown--high speed aircraft--185 km (100 nm) block distance.

Two conceptual engine designs were prepared based on the above cycle and incorporating the candidate STAT advanced technologies. The resultant engines are representative of those with performance levels anticipated for the 1990s' era commuter aircraft market. The engine configurations eventually selected to achieve those performance levels, however, could depart considerably from those described herein.

The advanced STAT engines are similar in that both have a single-spool gasifier and a free-power turbine with front drive. The smaller engine has an axial-centrifugal compressor with five axial stages. The final stage features a hybrid centrifugal impeller with a cast flow-path ring bonded to a forged hub by the hot isostatic press (HIP) process. The larger engine uses an all-axial compressor with nine stages. Airflow for the two engines is 6.35 and 11.34 kg/s (14 and 25 pps), respectively. Both engines have two-stage gasifier turbines employing impingement cooling in both stages plus the first-vane stage. Both turbines have hybrid turbine wheels with cast airfoil rings that are diffusion bonded to powdered metal hubs. Abrasive blade tips and abradable coatings are used to provide minimum running clearance in the turbines. Turbine shafts are fabricated of borsic-titanium matrix composite material for required shaft stability with simple rotor support. The combustors are transpiration cooled for improved temperature profile and increased hot section life.

The 19.1:1 reduction gear for the smaller engine is a close-coupled star/planetary gear system mounted concentrically on the front of the power section. The larger engine has an offset-type reduction gear with a dual compound idler gear train designed for a 10.4:1 reduction ratio. This reduction gear is connected to the power output shaft via a mechanical torquemeter. Both reduction gear cases employ a composite material.

The control and fuel systems for both STAT engines are configured to use an advanced digital electronic controller. An engine condition monitoring system is employed to gather data for component life usage and to assist in fault detection and isolation procedures so that corrective action can be planned to prevent failures. Modular construction is employed for both engines.

A benefit analysis was conducted comparing the advanced STAT engines with current technology engines as well as with hypothetical 1985 technology derivative engines. In the high speed airplane, the derivative engine was 3 to 4% better in DOC than the current technology engine for the average block distance of 185.2 km (100 nm). Also, the advanced STAT engine improved 15 to 16% in DOC compared to current technology. These results apply for fuel costs from \$0.264 to \$0.396/1 (\$1.00 to \$1.50/gal). Fuel consumption was reduced 11% with the derivative engine and 23% for the advanced STAT engine. Corresponding benefits in the low speed airplane were similar but with slightly lower improvement percentages, i.e., 13 to 14% improvement in DOC over current technology and 20% reduction in fuel consumption for the advanced STAT engine.

Research and development programs necessary to advance the state of the art for engine components, and an experimental engine program to provide basic operating data on the advanced technology elements and design features, were planned to achieve a 1988 date for readiness to release for full commercial development.

INTRODUCTION

The increasing need for newer, more efficient aircraft for specialized short-haul sectors of the transportation market has been generated by unprecedented growth in commuter traffic operations. Improved accommodations for the passenger and his baggage, similar to those of the major trunk line operations, are needed. Ride quality and sound levels that are customary for large, long-range aircraft are desired. To accomplish these goals requires a small commuter transport that is efficient in terms of fuel usage. The need for reduced fuel consumption has never been greater because of its rapidly rising price and decreasing availability. Reduced maintenance, which contributes significantly to lower operating costs and improved on-time performance, is also an important objective.

NASA LeRC sponsored the Propulsion Study for STAT reported herein. This study attempts to identify the advanced propulsion technology that is appropriate for the smaller short-haul type of aircraft. "Appropriate" technology is that which results in reductions in DOC, fuel usage, or total cost of ownership (TCO) size, cost, and maintenance cost for the power output required.

The study was divided into three basic tasks:

- Task I--Baseline Airplane, Engine, and Mission Definitions
- Task II--Advanced Technology Identification and Evaluation
- Task III--Recommendations for Future Research

Note: "Engine" as used in this report denotes power section plus reduction gearbox. Also note specific fuel consumption (SFC) is equivalent to brake specific fuel consumption (BSFC).

BASELINE AIRPLANE AND MISSIONS

The objective of Task I was to define two 50-passenger baseline commuter aircraft representing technology for 1980 initial operation. One of these aircraft was low speed, but with at least 463 km/h (250 kt) indicated air speed (IAS) capability at 3048 m (10,000 ft) altitude, and the other high speed with 0.7-M_N cruise capability. The airplanes use turboprop propulsion systems, also of 1980 initial operation capability (IOC) technology.

Reference aircraft data conforming to NASA aircraft and mission specifications and guidelines for economic calculations was obtained. Low speed aircraft data was provided by NASA Ames. The high speed aircraft information was furnished by Lockheed California Company (LCC), and was generated as a part of their STAT contract with NASA Ames. Details of the airplane requirements and the configuration will be discussed in the following paragraphs.

Hypothetical turboprop engines representing existing modern production technology 1980 IOC, along with appropriate scaling relationships over the horsepower range of interest are also defined and discussed in the subsequent section on Baseline Engines.

A computerized aircraft sizing, mission, and economic model was developed by DDA to accept the reference aircraft data provided by NASA Ames and LCC. The model is described, and the aircraft system sensitivity to engine parameters is presented in paragraphs on Aircraft Sizing and Cost Model, and Aircraft System Sensitivity to Engine Parameters.

MISSION REQUIREMENTS

The baseline missions conform to the following performance requirements:

- o Full design payload is carried over a range of 1111 km (600 nm) with IFR reserves for a 185.2 km (100 nm) alternate, and 45 min at maximum endurance power at 3048 m (10,000 ft) altitude.
- o Field length is limited to 1219 m (4,000 ft) for a hot day 306 k (90°F) at sea level, per FAR 25.
- o Aircraft meets current FAR 36 Stage 3 noise limits, minus 8 EPNdB at all measurement locations.
- o Cruise speed capability is at least 463 km/h (250 kt) indicated airspeed at 1829-3048 m (6,000-10,000 ft) altitudes, standard day conditions for the low speed airplane, and 0.7 M_N for the high speed airplane.
- o A terminal area speed capability is at least 334 km/h (180 kt) indicated airspeed with gear and flaps extended in order to stay with large jet aircraft.
- o Stall speed is less than 172 km/h (93 kt) in landing configuration at maximum landing weight in order to qualify for operations in Instrument Approach Category B aircraft requirements.

The mission profile shown in Figure 2 describes the features of the design and alternate missions used in the STAT study. The airplane objective was to carry 50 passengers, which was equated to 4536 kg (10,000 lbm) payload. Basis for specification of the mission segments is described herein.

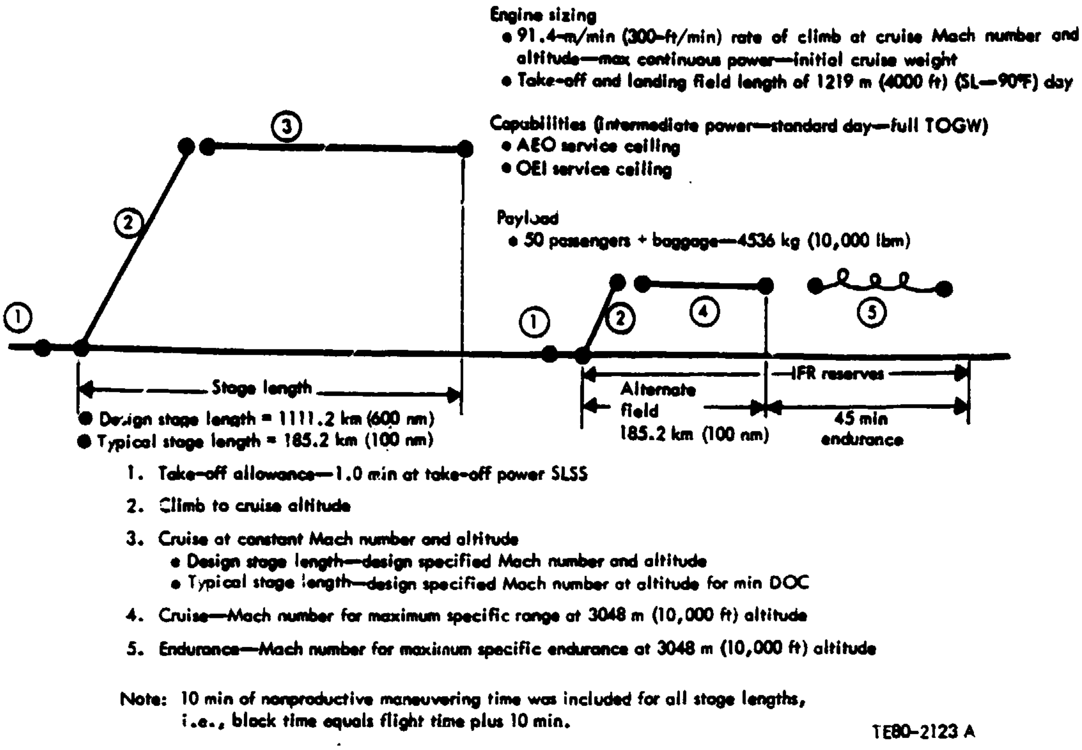


Figure 2. - Mission requirements.

A fuel allowance of 1 minute at maximum rated power was selected as representative of fuel usage for the take-off. All descents were assumed zero fuel and zero time segments.

The climb path was important in establishing mission fuel usage, block velocity, and block time, because climb distance constituted a substantial part of the stage length. Its selection has a direct bearing on DOC. Accordingly, a study was done to select a suitable climb path for each reference aircraft. Several constant IAS climbs were flown with each aircraft, and the resulting mission capabilities and DOC were examined. Climb speed could be no higher than 463 km/h (250 kt) IAS below 3048 m (10,000 ft) altitude. The climb speed selected for the LCC aircraft was a constant 417 km/h (225 kt) IAS, and for the NASA Ames aircraft, a 370 km/h (200 kt) IAS constant climb speed. These climb speed selections contributed a substantial DOC saving, and also provided terminal Mach numbers at the cruise altitude that closely matched the desired cruise Mach numbers.

The design mission stage length (climb plus cruise) was specified at 1111 km (600 nm) by NASA to represent typical commuter requirements. This range constituted the design range for sizing the aircraft. An alternate stage length of 182 km (100 nm) was used to examine the DOC implications of STAT engine technology with the sized airplane. The alternate stage length mission was exercised with full fuel load and payload at take-off; i.e., at design TOGW. The cruise altitude for the alternate stage length was optimized to obtain minimum DOC within the constraint that the cruise leg was at least half the climb-cruise stage length.

Additional mission segments were used to determine the fuel reserves necessary to meet the NASA specified alternate field capability. They were a take-off, climb, and IFR cruise at 3048 m (10,000 ft) altitude, maximum range speed for 182 km (100 nm), and a 45-min endurance at the same altitude at maximum endurance speed. (See details in Figure 2.)

Engine sizing conditions specified for the study were at take-off and the initial cruise point. A take-off and landing field length limit of 1219 m (4000 ft) was required on a +32°C (+90°F) day, and a 1.52 m/s (300 ft/min) minimum rate-of-climb was required at the initial cruise point with the engine at maximum continuous power.

BASELINE AIRPLANES

The baseline aircraft were required to utilize scaled versions of existing turboprop engines or notational turboprop engines representing existing, modern designs. They were also required to provide the following accommodation features:

	SI Units	Customary Units
o Weight per passenger plus baggage	90.7 kg	200 lbm
o Weight per crew member	90.7 kg	200 lbm
o Weight per flight attendant	59.0 kg	130 lbm
o Minimum interior aisle height	1.9 m	6 ft
o Minimum seat pitch	813 mm	32 in.
o Minimum seat width between armrests	457 mm	18 in.
o Minimum aisle width	457 mm	18 in.
o Preloaded baggage storage volume per passenger	0.14 m^3	5 ft^3
o Carry-on baggage storage volume per passenger	508 mm x 508 mm x 279 mm	20 in. x 20 in. x 11 in.
o Garment hanging storage width per passenger	20 mm	0.8 in.
o One lavoratory	- -	- -
o Minimum cabin pressurization	34.5 kPa	5 psi
o Maximum cabin interior noise level less than 85 db OASPL	- -	- -
o Speech interference level of less than 65 db	- -	- -

Airframe design life was required to be at least 30,000 h with 60,000 take-off-land cycles.

Physical characteristics of the low speed Ames aircraft and the high speed LCC aircraft are presented in Table I.

TABLE I. - BASELINE AIRCRAFT PHYSICAL CHARACTERISTICS

	Low speed	High speed
Data source	NASA-Ames	LCC
Design TOGW--kg (lbm)	17,927 (39,523)	18,291 (40,325)
Payload--kg (lbm)	4536 (10,000)	4536 (10,000)
Fuel load--kg (lbm)	1731 (3817)	1905 (4200)
OEW--kg (lbm)	11,660 (25,706)	11,850 (26,125)
Wing		
Span--m (ft)	25.4 (83.2)	21.6 (71.0)
Area--m^2 (ft^2)	61.19 (658.7)	46.8 (504)
Taper	0.3	0.3
1/4 C sweep--deg	5.0	5.38
AR	10.5	10.0
Loading--kPa (lb/ft^2)	2.87 (60.0)	3.83 (80.0)
Fuselage		
Length--m (ft)	22.8 (74.8)	22.8 (74.7)
Diameter--m (ft)	2.74 (9.00)	2.89 (9.50)
Horizontal tail		
Span--m (ft)	11.38 (37.35)	7.181 (23.56)
Area--m^2 (ft^2)	25.92 (279.0)	7.733 (83.24)
Vertical tail		
Span--m (ft)	6.300 (20.67)	3.514 (11.53)
Area--m^2 (ft^2)	24.81 (267.0)	6.172 (66.43)
Propeller		
No. blades	4	4
Diameter--m (ft)	4.51 (14.8)	3.66 (12.0)

Performance capability, aerodynamics, and economic characteristics of both airplanes as used in this study are presented in Table II. This data was obtained from the results of the STAT studies at NASA-Ames and Lockheed where engines of the appropriate power class were scaled to specific propulsion requirements.

Figure 3 shows the relationship of the baseline airplanes' characteristics to those of several contemporary transport aircraft types. The study vehicles are appropriately placed at the high speed border of the existing commuter equipment, and also provide a higher speed option that will elevate commuter equipment into a performance category competitive with corporate jet aircraft.

TABLE II. - BASELINE AIRCRAFT AERODYNAMIC AND ECONOMIC CHARACTERISTICS

	Low speed	High speed
Data source	NASA-Ames	LCC
Design range--km (nm)	1111 (600)	1111 (600)
Power per engine--kW (shp)	1767 (2369)	3573 (4792)
Cruise Mach number	0.47	0.7
Initial cruise altitude--m (ft)	6096 (20,000)	10,668 (35,000)
L/D at initial cruise	14.9	18.8
Lift coefficient at cruise	0.45	0.49
Engine-out service ceiling--m (ft)	4998.7 (16,400)	7132.3 (23,400)
Take-off distance--m (ft)	1219 (4000)	1218 (3995)
Approach speed--km/h (kt) EAS	209 (113)	204 (110)
Landing distance--m (ft)	1219 (4000)	1218 (3997)
Aircraft flyaway cost--1979 dollars	4,650,000	4,400,000
Operating cost* at 1111 km (600 nm)	2.73 (5.06)	2.30 (4.26)
Operating cost* at 185 km (100 nm)	3.82 (7.08)	4.92 (9.11)

*Measured in ¢/seat km (¢/seat nm) with fuel cost = $0.264/l ($1.00/gal)

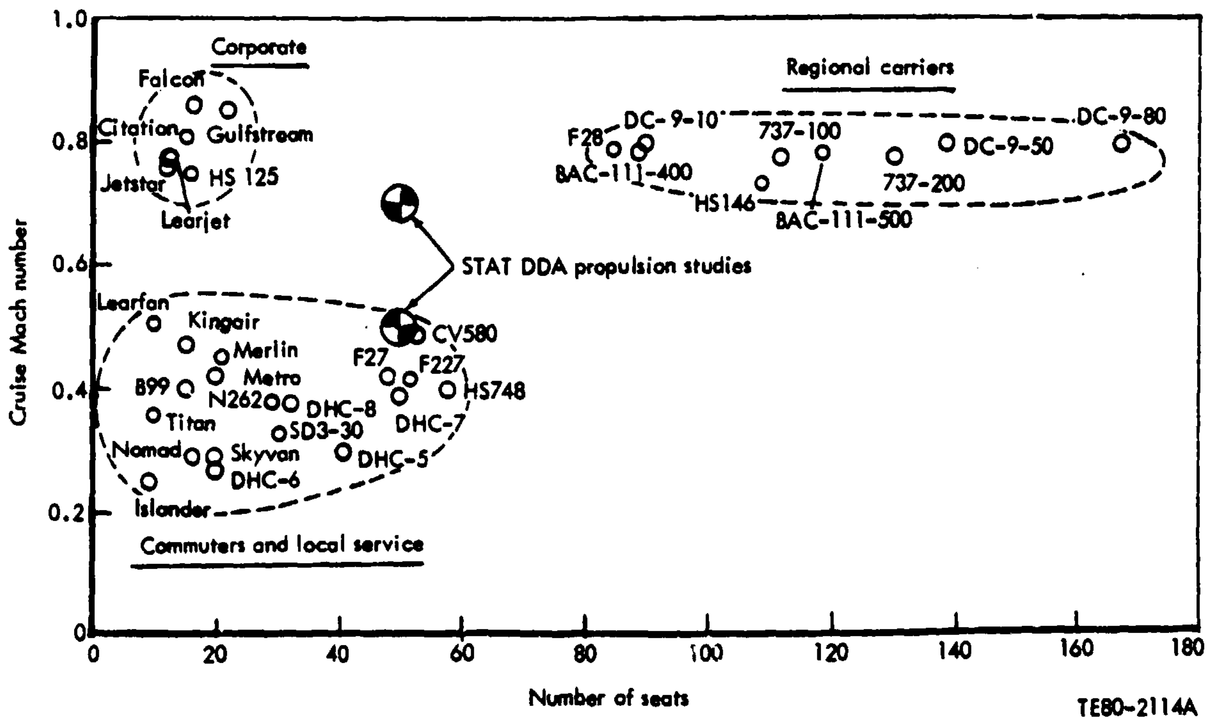


Figure 3. - STAT Propulsion study baseline aircraft.

BASELINE ENGINES

The most representative DDA current technology turboshaft engine over 746 kW (1000 hp), for conversion to turboprop applications, is the Model XT701-AD-700. This engine is a free turbine turboshaft that was developed through safety demonstration testing for the US Army's Heavy Lift Helicopter (HLH) program. This program was cancelled by the Army in December 1975, but had it continued through complete development for the HLH, it would have been operational in the early 1980s. DDA continued the engine development as a company-funded effort, directed toward industrial applications. The industrial version of the engine is designated the Model 570-K. It went into production in early 1979.

The turboprop engine that was derived from the Model XT701 power section is shown in Figure 4. It was designated the Model PD370-37 turboprop engine, and was derived during earlier Maritime Patrol Aircraft (MPA) engine studies for the Navy, where it also represented a low risk, current technology approach for application to that aircraft system. The speed reduction gearbox for the propeller was based on a standard T56 current production unit.

The Lockheed STAT short haul study was being conducted simultaneously with the advanced technology propulsion system studies at DDA, Garrett, and General Electric. To define an appropriate scalable baseline engine, and to ensure a degree of consistency with the engine technology studies, DDA, Garrett, and GE were asked by Lockheed to recommend a baseline engine for each of the Lockheed study aircraft and to provide scalable performance for the recommended engines.

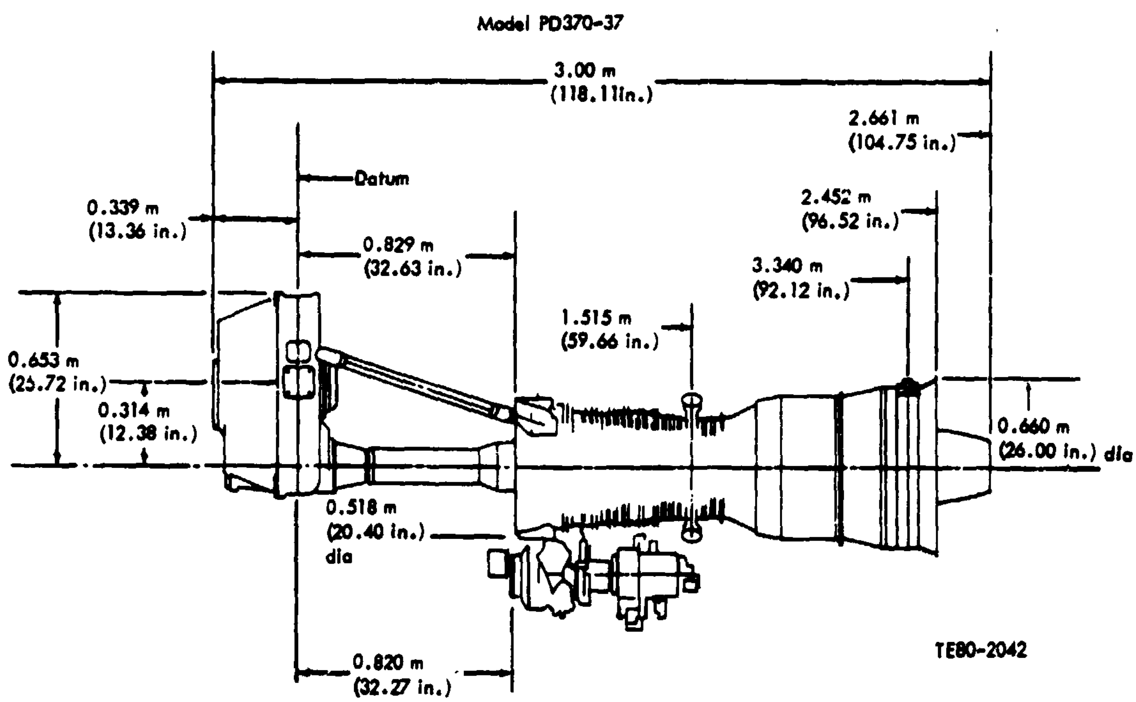


Figure 4. - Baseline turboprop engine.

DDA and the other engine manufacturers responded with the engines described in Table III. In general, the list represented the latest turboprop engine each manufacturer had to offer.

TABLE III. - ENGINE MANUFACTURER-RECOMMENDED BASELINE ENGINES

Engine manufacturer	Model engine	Rating-- kW (shp)	Basis	Comments
Allison	PD370-37	6301 (8450)	Proposed turboprop version of Model T701-AD-700 HLH engine developed through PPFRT (cancelled 1975). Now in stationary powerplant use.	Scaling corrections provided to scale engine to 1491 kW (2000 hp) range
Garrett	TPE331-11	745.7 (1000)	Production engine. Latest in T76 family.	Scaling corrections provided to scale engine to 2237 kW (3000 hp)
GE	CT7-2	1118.6 (1500)	Proposed turboprop version of T700 turboshaft engine (preliminary data).	
	CT64-820-4	2339 (3137)	Production engine (1960 vintage).	
P&W Canada	None			Only provided acquisition and maintenance cost estimates

Estimates of engine power class required for the 30 and 50 passenger baseline aircraft were 1491 kW (2000 hp) and 2983 kW (4000 hp), respectively. Since none of the recommended engines was of these sizes, scaling corrections were required to match engine power capability with aircraft thrust requirements. DDA and Garrett provided estimates for scaling corrections of the recommended powerplants.

Scaling equations that scaled characteristics down to engine sizes of approximately 1119 to 1491 kW (1500 to 2000 hp), were generated and keyed to the PD370-37. These equations represent technology trends and not scaling relations for a specific engine configuration.

The baseline scaled engine performance, weight, geometry, and cost data were based on a synthesis of the recommended engine data and scaling corrections. The following paragraphs describe the basis for establishing the baseline engine definitions.

Figure 5 presents the sea level static (SLS) uninstalled engine brake sfc for existing production turboprop engines as a function of rated horsepower. The performance level of the recommended engines is included. On the basis of this data, a baseline engine performance trend was established (i.e., dashed line on the figure). The sfc level of the engines in the two power classes are noted.

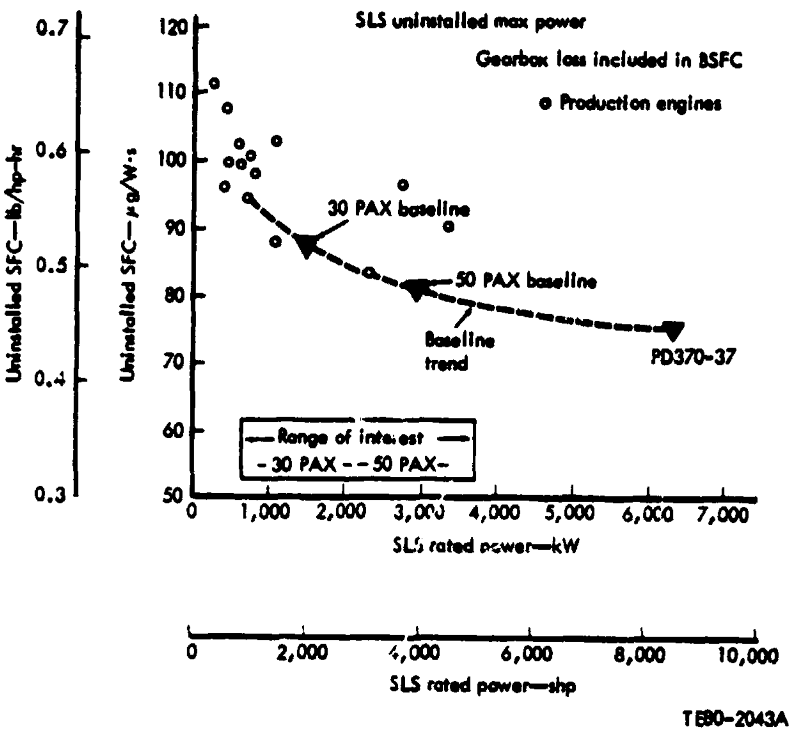


Figure 5. - SFC versus rated power.

Figure 6 presents the engine power-to-weight ratios for the baseline engines and for other production engines. The selected trend of horsepower to weight ratio with rated power, as well as the values chosen for the 30 and 50 PAX aircraft engine power classes, are noted.

Similarly, Figures 7 through 9 present the baseline trend established for engine length, overall envelope width, and overall envelope height, respectively, based upon the recommended and existing production engines. It is assumed that engine-mounted accessories are located in the horizontal plane.

DDA, Garrett, GE, and P&W of Canada were asked by Lockheed to define engine (including gearbox) acquisition and maintenance cost estimates (1979 dollars) as a function of engine-rated horsepower for engines representative of today's technology and design practices. Figure 10 presents the estimated acquisition cost in terms of dollars per horsepower. Ground rules included the assumptions that the gearbox price was included, and that the engine was mature. The recommended baseline trend is shown as the dashed line. Costs are assumed representative of OEM levels. Similarly, the burdened maintenance cost estimates and selected base-lines for the aircraft are shown in Figure 11.

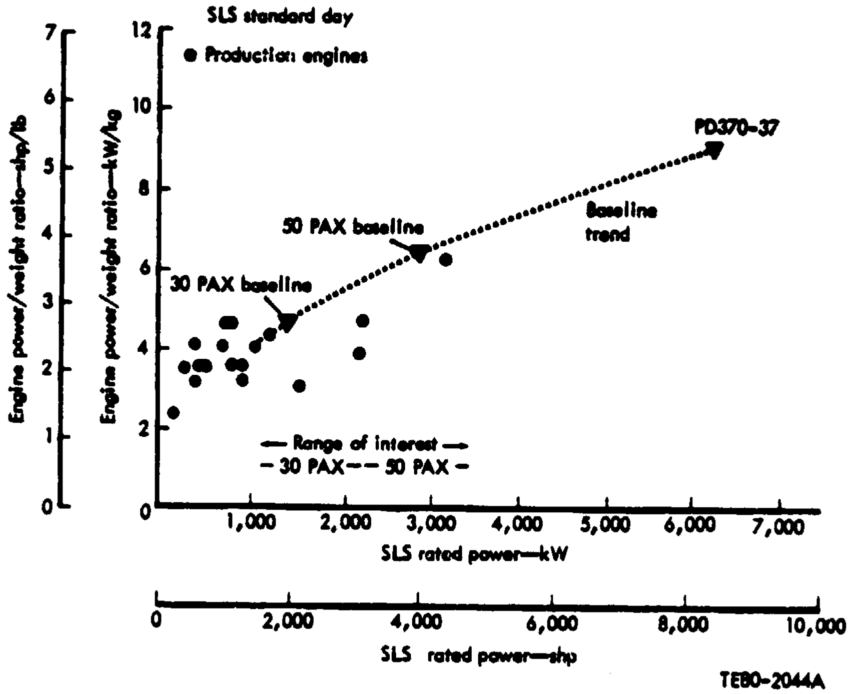


Figure 6. - Engine power-to-weight ratio versus rated power.

The variations in engine performance, weight, geometry, and economic trends with size are quantified in equation form, and are tabulated in Table IV. These equations were used to define the principal characteristics of existing turboprop propulsion systems in the 745.7- to 5965.6 kW (1000- to 8000 hp) range. These scaling data, applied to the Model PD370-37, were used to define baseline engine characteristics for evaluation of advanced propulsion system technologies.

AIRCRAFT SIZING AND COST MODEL

This section presents the following items:

- o Overview of the STAT mission analysis computer program
- o Engine/airframe sizing philosophy
- o Cost model
- o Results from Task I mission studies

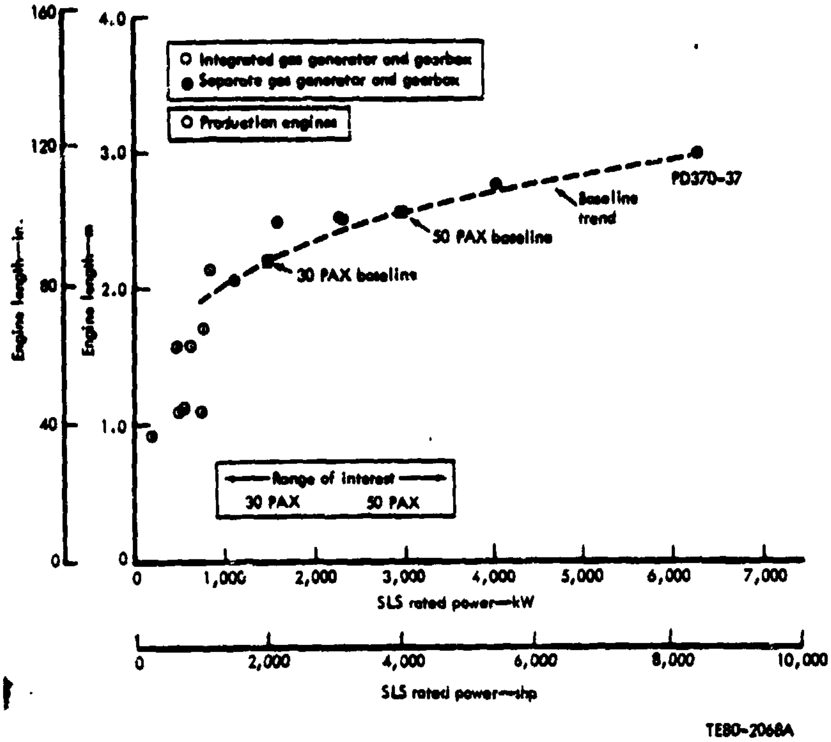


Figure 7. - Engine length comparison.

TABLE IV. - STAT SHORT HAUL BASELINE ENGINE MODEL

Synthesis of existing engine characteristics representative of today's technology and design practices.

Characteristic	SI Units	Customary Units
Engine BSFC--$\mu g/W \cdot s$ (lb/hr-hp) including gearbox loss	196.32 $kW^{0.11}$	1.200 $shp^{0.11}$
Weight--kg (lb) Engine	6.505 $kW^{0.54}$	12.24 $shp^{0.54}$

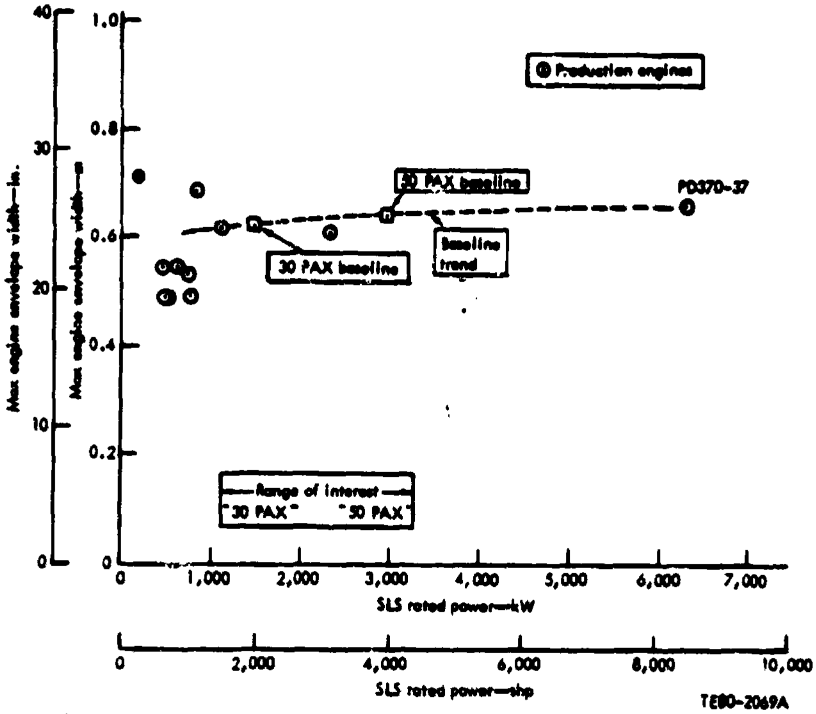


Figure 8. -- Engine max envelope width.

TABLE IV. (CONT)

Characteristic	SI Units	Customary Units
Geometry--m (in.)		
Length--prop mount flange to rear engine flange	0.4746 $kW^{0.21}$	17.57 $shp^{0.21}$
Max envelope height	0.10225 $kW^{0.26}$	3.73 $shp^{0.26}$
Max envelope width	0.4680 $kW^{0.04}$	18.21 $shp^{0.04}$
OEM acquisition cost (1979 $)		
Engine	1463.8 $kW^{0.7}$	1192 $shp^{0.7}$
Maintenance cost (1979 $/flight hour)		
Engine	0.2949 $kW^{0.66}$	0.243 $shp^{0.66}$

shp--engine max rated power, SLS, standard day.
BSFC--engine brake specific fuel consumption at max rated power, SLS standard day. Assumes exhaust nozzle area sized for turboprop application.

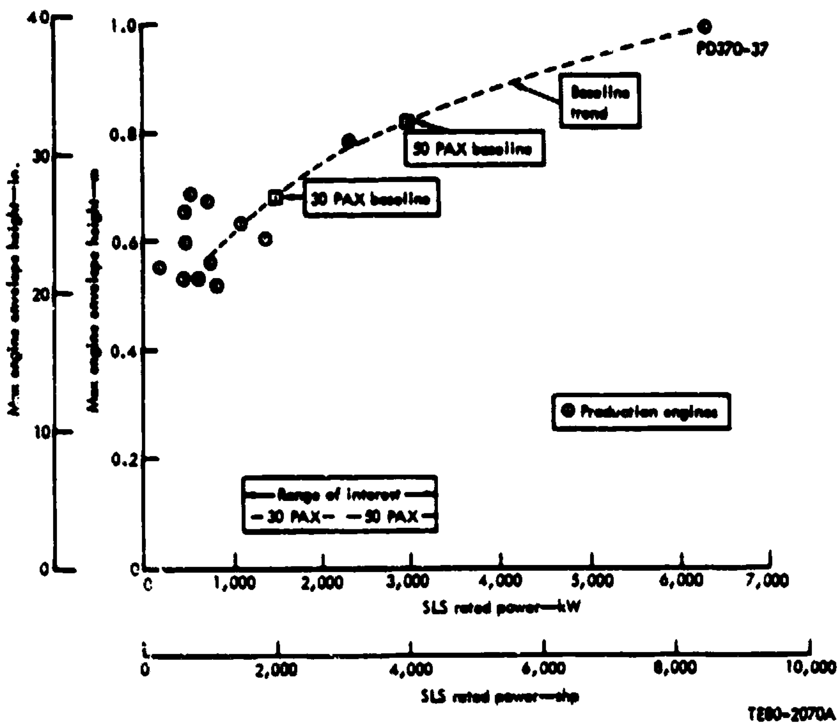


Figure 9. - Engine max envelope height.

Overview of STAT Mission Program

Input, major calculation functions, and output of the STAT mission program have been generalized in the block diagram shown in Figure 12. Mission requirements, aircraft, and engine data are used in the engine/airframe sizing calculations to determine the exact engine/aircraft size combination that will meet the specified mission requirements. The resultant aircraft, mission fuel and time data, plus input economic criteria are used in the cost routine to calculate the required cost parameters.

Engine/Airframe Sizing

A general flow diagram of the engine/airframe sizing procedure used in the STAT study is shown in Figure 13.

Definition of the mission requirements, baseline or reference aircraft characteristics, and unity size engine data constitutes the first step in the procedure. Mission range, speed, and payload must be specified along with the airframe critical propulsion requirements necessary to establish engine size. Baseline aircraft dimensions, geometry, component weight breakdown, aerodynamics, airframe cost factors, and installation information must also be specified. Unity installed engine performance is input for those altitudes, velocities, and power settings needed to calculate engine size and mission fuel usage. Engine dimensions and weight data are also required.

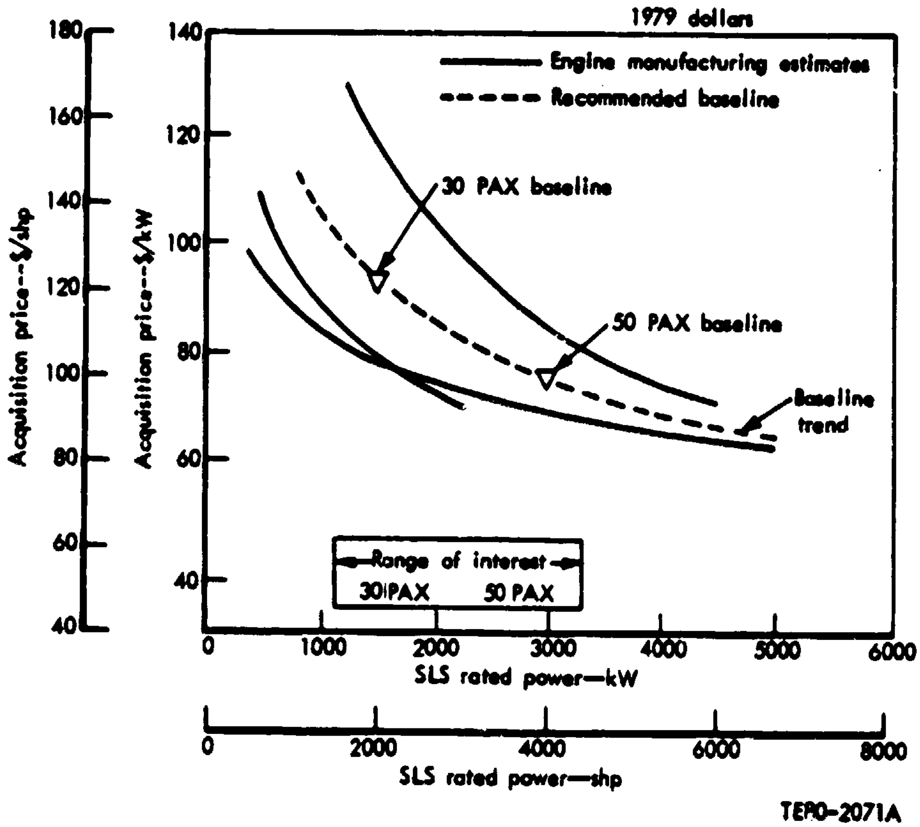


Figure 10. - Engine acquisition cost versus rated power.

Uninstalled engine performance is adjusted to reflect appropriate installation effects due to inlet characteristics, aircraft services, and propeller characteristics. The installation factors used to obtain installed engine performance are listed in Table V. This table lists the propeller efficiencies (ηp) used for take-off, climb, and cruise power conditions for all engines evaluated in the STAT study. These propeller efficiencies are typical of the Lockheed Electra/P3 propeller performance. Inlet recovery, gearbox loss, bleed, and power extraction factors were also applied to all engines.

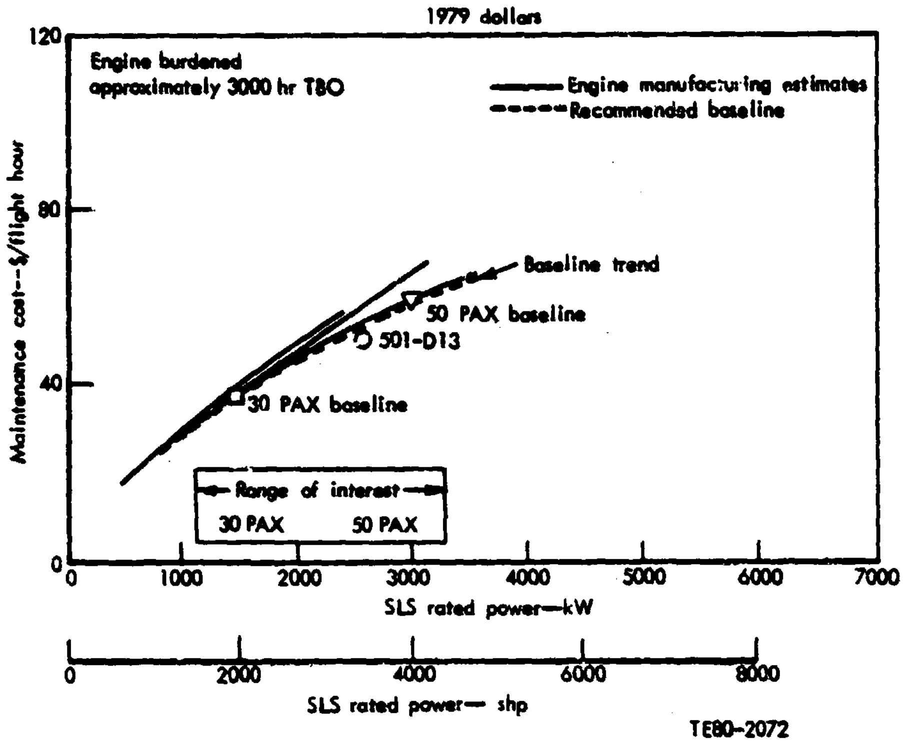


Figure 11. - Engine maintenance cost versus rated power.

TABLE V. - PROPULSION INSTALLATION CRITERIA

Efficiencies (typical Electra/P3 propeller)

Propeller	
o Take-off power--thrust-to-static-shp ratio	2.50
ηp at 0.1 M_N	0.26
ηp at 0.2 M_N	0.57
ηp at 0.3 M_N	0.73
o Climb power, ηp	0.84
o Cruise, ηp (0.7 M_N cruise)	0.81
o Cruise, ηp (0.5 M_N cruise)	0.90
Installed performance	
o Inlet recovery	1.00
o Gearbox power loss--%	2
o Customer bleed-ppm	0
o Customer power extraction--kw (shp)	55.9 (75)/engine

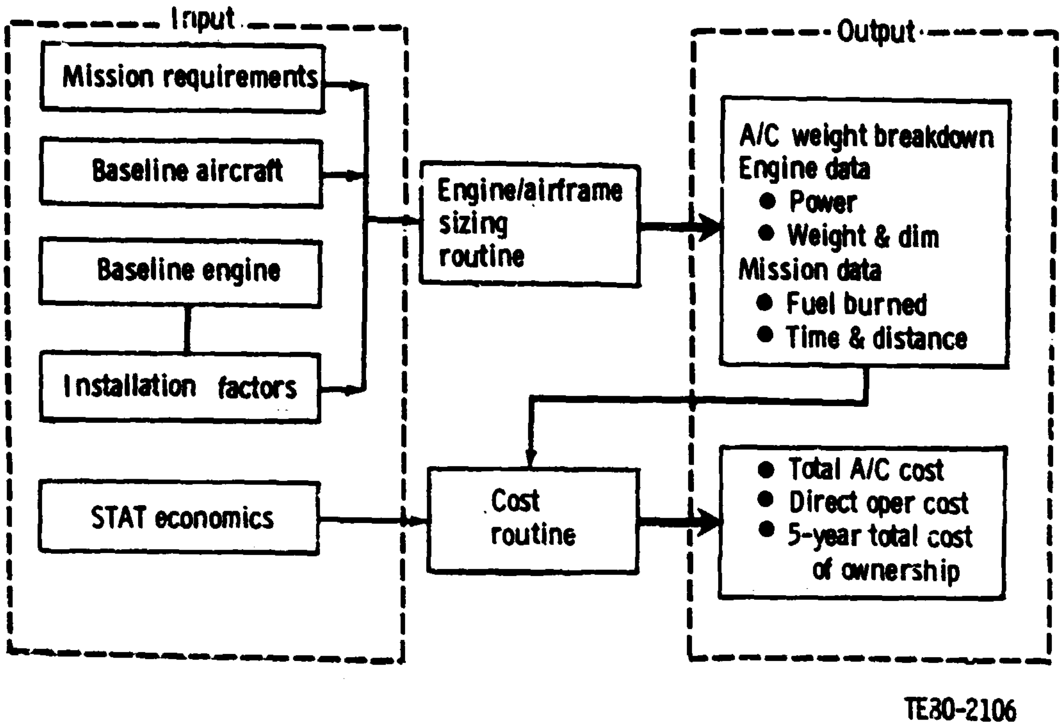


Figure 12. - STAT mission program.

The engine/airframe sizing is initiated with the selection of a "start point" trial vehicle gross weight. The engines are scaled to meet the most critical power requirement. Scaling equations developed by Lockheed for engine technology trending were utilized for all engines evaluated in the DDA model. Baseline aircraft dimensions, geometry, and component weights are then scaled to correspond to the trial gross weight. The power or drag is adjusted as the vehicle geometry varies from the baseline as driven by propulsion influences, thus requiring iteration of the engine sizing power or thrust. It is noted that the resultant characteristics are similar to the reference aircraft, but modified for scale effects. The fuel available is calculated for this trial gross weight and compared to the fuel required to perform the design mission flight profile. If the fuels are not equal, a new trial gross weight is selected and the process repeated. Upon convergence of the fuel weight iteration (fuel available = fuel required), a complete description of the scaled aircraft/engine combination is obtained. Dimensions, geometry, weight breakdown, engine sizing data, mission time, power setting, and fuel breakdowns are provided for on-line cost calculations for the complete aircraft including the engine and airframe.

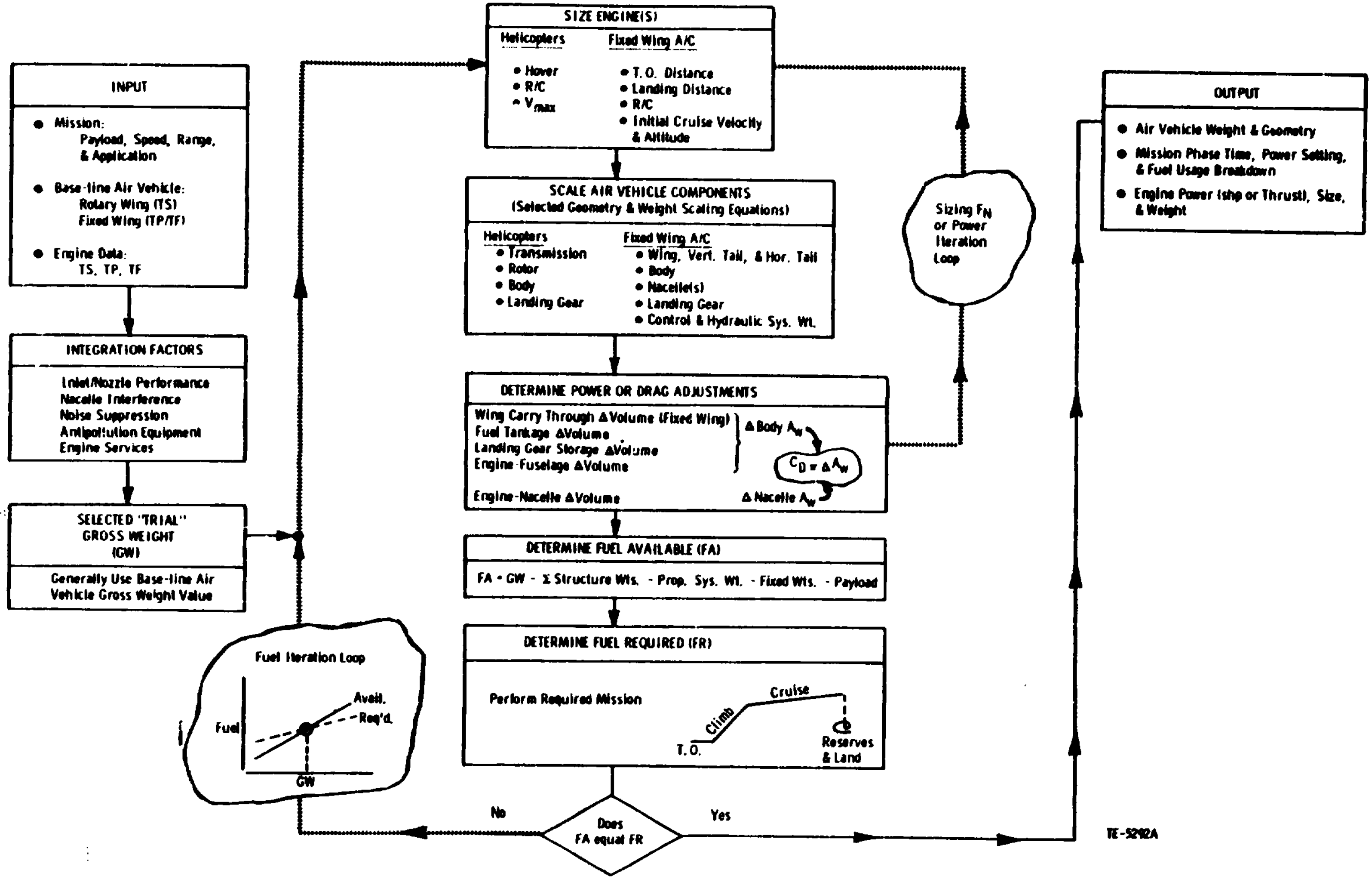


Figure 13. - Engine/airframe sizing flow diagram.

STAT Cost Model

The previously discussed engine/aircraft results, plus the cost assumptions listed in Table VI, are used with the cost equations presented in this section to obtain total aircraft cost (TAC), direct operating cost (DOC), and 5-year total cost of ownership (TCO).

TABLE VI. - COST ASSUMPTIONS

Base year dollars	1979
Fuel cost--\$/L (\$/gal)	0.264 (1.00) and 0.396 (1.50)
Utilization--h/y	2800
Insurance--%/y	1.5
Depreciation	Total flyaway price plus spares to 15%, residual in 12 years
Spares investment--%	
Engine	30
Aircraft less engine	6
Crew cost--\$/(block hour x number of seats)	2.50

Total Aircraft Cost (TAC)

TAC is calculated as follows:

- o Engine acquisition price = 1.50 x OEM price
- o Propeller acquisition price
  = \$2653.09 x Dp^2 x $(Ep/Dp^2)^{0.12}$--SI units
  = \$350.11 x Dp^2 x $(Ep/Dp^2)^{0.12}$--customary units
  where: Dp = propeller diameter---m (ft)
  Ep = engine power---------kW (shp)
- o Airframe acquisition price
  = FC + PC + (AFSC x AFWT)
  where: FC = Fixed costs
  (Low speed A/C) FC = \$1,927,000
  (High speed A/C) FC = \$1,734,000
  PC = Procurement costs (logistics support, product development, etc.)
  (Low & High speed A/C) PC = \$503,000
  AFSC = Airframe specific cost
  (Low speed A/C) AFSC = \$254/kg (\$115/lbm)
  (High speed A/C) AFSC = \$196/kg (\$89/lbm)
  AFWT = Airframe weights (fuselage, wing, vertical tail, horizontal tail, landing gear, and nacelle)

TAC = (Engine + Propeller + Airframe) x 1.10
(Flyaway Price) (Acquisition Price) (Mark up)

Direct Operating Cost (DOC)

The elements of DOC are fuel and oil, insurance, depreciation, crew cost, airframe maintenance, and engine maintenance cost. These items are calculated as follows:

- o Fuel and oil) These elements were calculated
- o Insurance) using December 1967 ATA
- o Depreciation) Standard Methods

- o Crew Cost = \$2.5/(block hour x number of seats)

- o Airframe maintenance costs--\$/block hour
  Maintenance cost = 0.3685 x (Empty Wt.--kg)$^{0.569}$ x AF
  Maintenance cost = 0.235 x (EMpty Wt.--lbm)$^{0.569}$ x AF

Where: AF = Adjustment factor
(Low speed A/C) AF = 1.0
(High speed A/C) AF = No. of departures/No. of departures for the design stage length mission.

Note: No.of departures = Utilization/Block time
For design stage length mission - - - - - AF = 1.0
For alternate stage length missions - - - AF > 1.0
i.e., stage lengths less than 1111 km (600 NM)

- o Engine maintenance costs -- \$/flight hour

engine maintenance cost = \$0.295 x (power--kW)$^{0.66}$
engine maintenance cost = \$0.243 x (power--shp)$^{0.66}$
propeller maintenance cost = \$4.15/flight hour

Further breakdown of both airframe and engine maintenance cost used in the STAT cost model is shown in Table VII.

TABLE VII. - MAINTENANCE COST BREAKDOWN

Airframe Maintenance:
Direct material cost = 0.5 x direct airframe maintenance cost
Direct labor cost = 0.5 x direct airframe maintenance cost
Burden = 0.8 x direct labor cost

Engine (power section + gearbox + propeller) maintenance:
Direct material cost = 0.8 x direct engine maintenance cost
Direct labor cost = 0.2 x direct engine maintenance cost
Burden = 0.8 x direct labor cost

Total Cost of Ownership (TCO)

The total cost of ownership was defined by DDA to be the cost of financing the purchase of the aircraft plus the cost of operating these aircraft over a period of 5 years. The finance rate was assumed to be 10%.

TCO = 5-y cost of financing aircraft purchase + 5-y cost of operation

$$TCO = (1.10^5 - 1.00) \times TAC + DOC \times CF \times VB \times U \times 5\ y$$

where:
- TAC = Total aircraft cost (flyaway)--$
- DOC = Direct operating cost--¢/seat km (seat nm)
- CF = Conversion factor (50 seats/100 ¢)
- VB = Block velocity--km/h (nm/h)
- U = Utilization rate--2800 h/y

Task I Mision Study Results

Results from studies completed during the Task I effort are presented as follows:

- o Cruise altitude selection
- o Climb path schedule
- o Rate of climb capabilities

Cruise Altitude Selection

The cruise altitude and Mach number used by DDA for the design stage length, 1111.2-km (600 nm) mission were fixed at values approximate to those specified for the baseline aircraft.

	Design Cruise Altitude (ft)	Design Cruise Velocity (M_n)
Low Speed Aircraft	20000	0.47
High Speed Aircraft	35000	0.70

The cruise altitudes for the off-design stage length missions were selected to minimize DOC at the aircraft design cruise Mach number.

Figure 14 shows a typical plot of altitude versus DOC for the high and low speed aircraft flying the 185.2 km (100 nm) stage length mission. This figure indicates the minimum DOC altitude for the low speed aircraft to be at 3048 m (10,000 ft) and at 5791 m (19,000 ft) for the high speed aircraft.

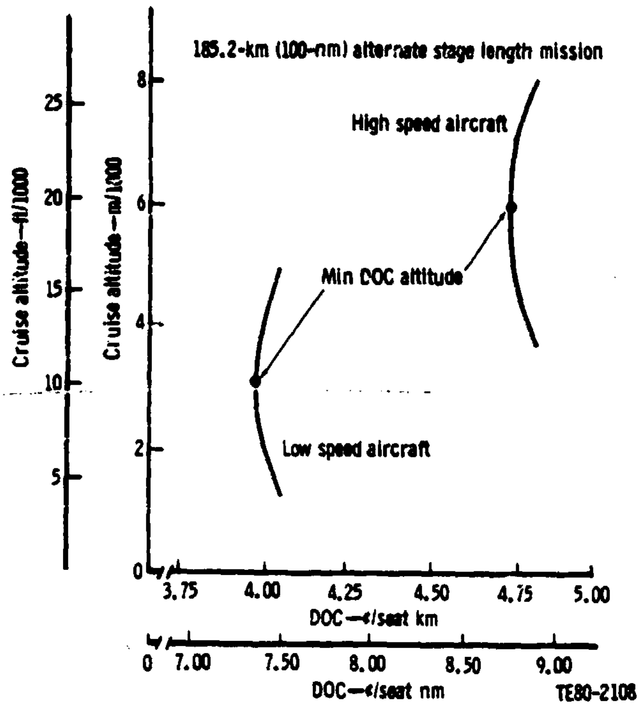


Figure 14. - Cruise altitude selection.

Climb Path Schedule

Because of the importance of the climb path in establishing mission fuel usage, block velocity, etc., a study was completed to select the most suitable climb path for both aircraft types. Several climb paths of constant indicated air speed (IAS) were flown in the design mission (climb to cruise altitude plus climb to alternate field cruise altitude) with a current technology engine. The resulting aircraft were then flown in the 182.5 km (100 nm) off-design mission with the constant IAS climb that was used to establish the design aircraft. The 185.2 km (100 nm) DOC results from each climb velocity path are shown for the high speed aircraft in Figure 15, and for the low speed aircraft in Figure 16.

It is noted that the initial or lowest climb velocity is approximate to the path that would be flown for a maximum rate of climb schedule. Also, the climb velocity was not allowed to exceed 463 km/h (250 kt) IAS below an altitude of 3048 m (10,000 ft). The climb schedule selected for the high speed aircraft was a constant 417 km/h (225 kt) IAS path and for the low speed aircraft a 370.4 km/h (200 kt) IAS path. The selected climb paths were incorporated into the mission and utilized to evaluate each engine technology. They provide a substantial portion of the DOC reduction indicated for the range of

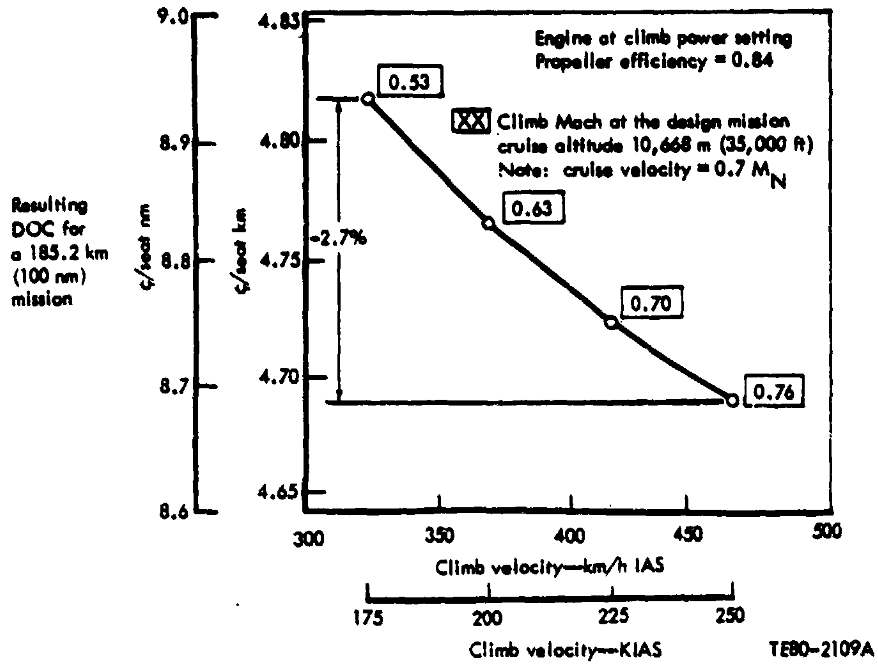


Figure 15. - Climb study results--LCC aircraft.

constant IAS climb velocities evaluated, plus terminal climb Mach numbers that closely match the cruise Mach number for the design mission. However, in the case of the off-design missions, this final climb/cruise Mach number match is not maintained due to the optimization of the cruise altitude to minimize DOC.

Rate of Climb Capabilities

The rate of climb capabilities for standard day, all engine operative (AEO) and one engine inoperative (OEI) maximum rate of climb at sea level, and aircraft service ceilings for the current technology engine-powered low and high speed aircraft are shown in Table VIII. Note that the power loading for the high speed aircraft is approximately 70% higher than that for the low speed aircraft.

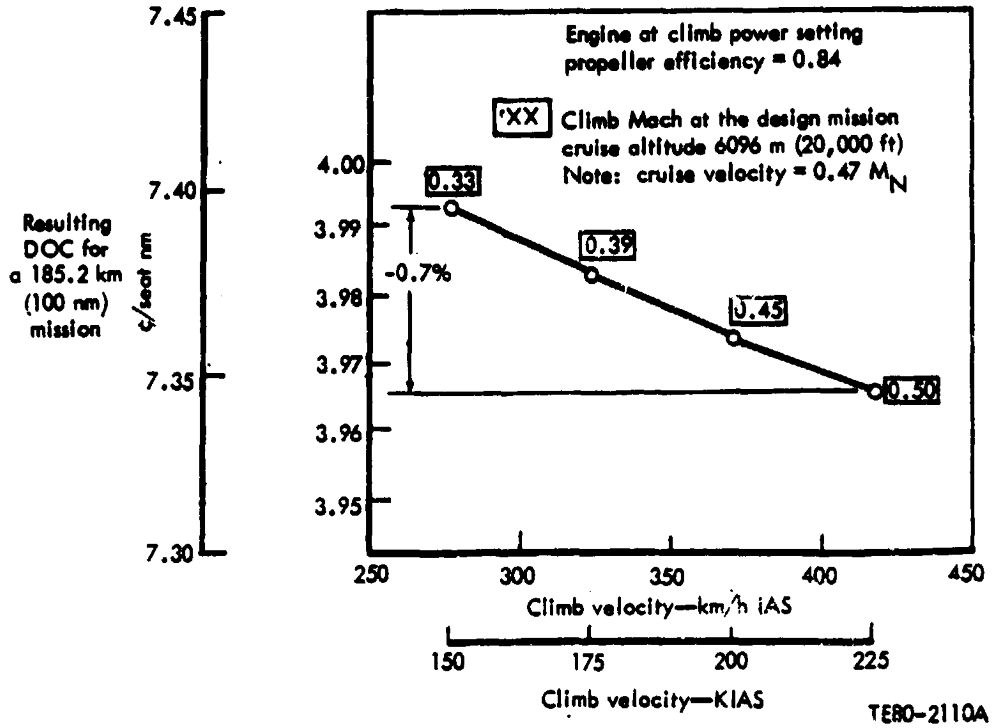


Figure 16. - Climb study results—Ames aircraft.

TABLE VIII - RATE OF CLIMB CAPABILITIES

- o Current technology engine (Intermediate power)
- o Standard day conditions
- o Full design takeoff gross weight

	Low speed aircraft	High speed aircraft
All Engine Operative (AEO)		
o SL maximum rate of climb—m/min(fpm)	884 (2900)	1707 (5600)
o Service ceiling—m (ft)	8534 (28000)	12192 (40000)
One Engine Inoperative (OEI)		
o SL maximum rate of climb—m/min (fpm)	299 (980)	674 (2210)
o Service ceiling—m (ft)	4877 (16000)	7010 (23000)

AIRCRAFT SYSTEM SENSITIVITY TO ENGINE PARAMETERS

This paragraph presents the sensitivity data developed in Task I for the high and low speed study aircraft powered by a current technology engine.

Table IX summarizes the percentage changes in direct operating cost values (185.2 km (100 nm) stage length mission) for 10% improvements in each of the following engine parameters:

TABLE IX. - SUMMARY OF DDA SENSITIVITY RESULTS

Engine Parameter (10% Improvement)	Reduction in DOC*, % High speed aircraft	Low speed aircraft
Overall sfc	3.5	4.12
Weight (dry)	0.30	0.28
OEM price	0.40	0.41
Price (plus maintenance parts)	1.29	1.10
Maintenance	1.28	1.00
Maintenance labor only	0.40	0.31
Max envelope length	0.28	0.19
Max envelope height	0.41	0.27

* 185.2 km (100 nm) alternate stage length mission and fuel cost = $0.264/L ($1.00/gal).

The relative importance of engine sfc, weight, and cost can be determined from the data presented in Table IX, and is illustrated in the following equations which show the percentage improvement in each parameter required to obtain a 1% reduction in DOC:

High speed aircraft --

1% DOC = 2.8% engine sfc = 33.3% engine weight = 25% engine cost

Low speed aircraft --

1% DOC = 2.4% engine sfc = 35.7% engine weight = 24.4% engine cost

It is noted that baseline aircraft parameters and propulsion system weights, dimensions, OEM prices, and maintenance costs are listed for reference purposes in Table X.

Additional data from the sensitivity studies of the current technology engine (CTE)-powered high and low speed aircraft configuration are presented in Tables XI through XIV. Each table lists changes in gross weight, empty weight, aircraft acquisition cost, block fuel, and DOC resulting from 10% improvements in each parameter listed.

TABLE X. - BASELINE DATA--DDA STUDY AIRCRAFT

	High speed aircraft	Low speed aircraft
TOGW--kg (lbm)	18,299 (40,343)	18,938 (41,730)
Engine take-off rating at SLSS--kW (shp)	13,531 (4735)	2139 (2868)
Aircraft flyaway cost--1979$	5,456,000	4,836,000
Weights (dry) per engine--kg (lbm)		
Power section	367 (808)	279 (616)
Gearbox	166 (367)	127 (280)
Total	533 (1175)	406 (896)
Propeller	420 (926)	254 (561)
Dimensions		
Engine max envelope length--m (in.)	2.66 (104.6)	2.39 (94.1)
Engine max envelope height--m (in.)	0.85 (33.3)	0.74 (29.2)
Propeller diameter--m (ft)	3.66 (12.0)	2.83 (9.3)
OEM price per engine--1979 $		
Power section	308,213	216,979
Gearbox	13,682	9,632
Total	321,895	226,611
Propeller	50,881	30,817
Maintenance (fully burdened) 1979 $/EFH		
Engine	64.75	46.50
Propeller	4.15	4.15

DOC Breakdown

Figures 17 and 18 present a breakdown of the engine-related DOC elements for each baseline aircraft in a pie graph format. These data are for the 185.2 km (100 nm) stage length mission and a fuel cost of $0.264/L ($1.00/gal). Figures 17 and 18 indicate the largest cost item to be fuel (30 to 34% of the total aircraft DOC) followed by engine maintenance (11 to 14% of the total aircraft DOC). In addition, Figures 17 and 18 show the effect of doubling fuel cost to $0.528/L ($2.00/gal). This increased fuel cost drives the engine related cost share from approximately 50 to 60% of total DOC.

Propeller Sensitivity Data

Table XV presents propeller sensitivity data determined from the previously discussed engine sensitivity results. This table shows the effect on DOC for the 0.7 Mach aircraft, of a 1% change in three propeller parameters. These data indicate propeller efficiency to be the most significant parameter.

TABLE XI. - ADDITIONAL SENSITIVITY DATA--HIGH SPEED AIRCRAFT (SI UNITS)

10% improvement in engine parameters	TOGW reduction--kg	Mfg empty weight reduction--kg	Flyaway cost reduction--1979 $	Block fuel reduction--kg		DOC reduction--¢/seat km ($0.264/L)	
				1111.2 km	185.2 km	1111.2 km	185.2 km
Overall sfc	298	82	27,129	127	41	0.084	0.166
Weight (dry)	156	146	13,269	6	2	0.007	0.014
OEM price	---	---	106,226	---	---	0.012	0.019
Maintenance (fully burdened)	---	---	---	---	---	0.040	0.060
Max envelope length	48	38	12,875	6	2	0.007	0.013
Max envelope height	68	54	18,462	9	3	0.010	0.019

Baseline Characteristics

TOGW = 18,299 kg
Mfg empty weight = 11,328 kg
Flyaway cost = $5,456,141

Block fuel (1111.2 km) = 1190 kg
Block fuel (185.2 km) = 380 kg

DOC (1111.2 km) = 2.310 ¢/seat km
DOC (185.2 km) = 4.726 ¢/seat km

TABLE XII. - ADDITIONAL SENSITIVITY DATA--HIGH SPEED AIRCRAFT (CUSTOMARY UNITS)

10% improvement in engine parameters	TOGW reduction--lbm	Mfg empty weight reduction--lbm	Flyaway cost reduction--1979 $	Block fuel reduction--lbm		DOC reduction--¢/seat nm ($1.00/gal)	
				600 nm	100 nm	600 nm	100 nm
Overall sfc	658	181	27,129	281	90	0.156	0.307
Weight (dry)	345	322	13,269	13	4	0.013	0.026
OEM price			106,226			0.023	0.035
Maintenance (fully burdened)					---	0.074	0.112
Max envelope length	105	83	12,875	14	4	0.013	0.025
Max envelope height	151	118	18,462	20	6	0.019	0.036

Baseline Characteristics

TOGW = 40,343 lbm
Mfg empty weight = 24,974 lbm
Flyaway cost = $5,456,141

Block fuel (600 nm) = 2623 lbm
Block fuel (100 nm) = 838 lbm

DOC (600 nm) = 4.278 ¢/seat nm
DOC (100 nm) = 8.752 ¢/seat nm

TABLE XIII. - ADDITIONAL SENSITIVITY DATA--LOW SPEED AIRCRAFT (SI UNITS)

10% improvement in engine parameters	TOGW reduction-- kg	Mfg empty weight reduction-- kg	Flyaway cost reduction-- 1979 $	Block fuel reduction-- kg		DOC reduction-- ¢/seat km ($0.264/L)	
				1111.2 km	185.2 km	1111.2 km	185.2 km
Overall sfc	359	125	32,025	161	41	0.109	0.164
Weight (dry)	134	125	11,250	7	2	0.008	0.011
OEM price	---	---	74,782	---	---	0.012	0.016
Maintenance (fully burdened)	---	---	---	---	---	0.038	0.040
Max envelope length	36	29	7,586	5	1	0.005	0.008
Max envelope height	50	41	10,743	7	2	0.008	0.011

Baseline Characteristics

TOGW = 18,928 kg
Mfg empty weight = 11,828 kg
Flyaway cost = $4,835,535

Block fuel (1111.2 km) = 1466 kg
Block fuel (185.2 km) = 369 kg

DOC (1111.2 km) = 2.907 ¢/seat km
DOC (185.2 km) = 3.972 ¢/seat km

TABLE XIV. - ADDITIONAL SENSITIVITY DATA--LOW SPEED AIRCRAFT (CUSTOMARY UNITS)

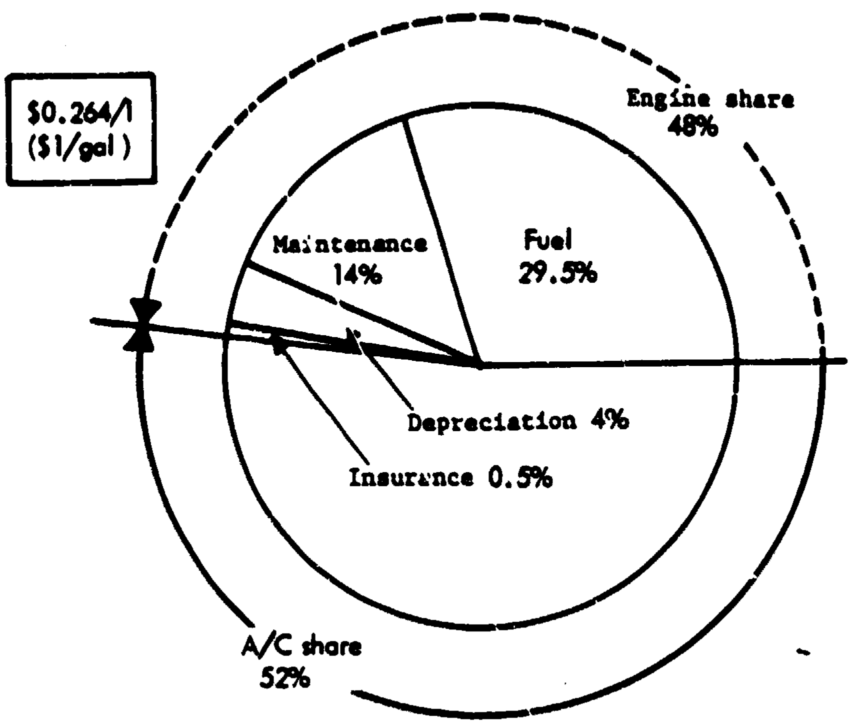
10% improvement in engine parameters	TOGW reduction-- lbm	Mfg empty weight reduction-- lbm	Flyaway cost reduction-- 1979 $	Block fuel reduction-- lbm		DOC reduction-- ¢/seat NM ($1.00/gal)	
				600 nm	100 nm	600 nm	100 nm
Overall sfc	792	275	32,025	355	91	0.202	0.304
Weight (dry)	296	275	11,250	15	4	0.015	0.021
OEM price	---	---	74,782	---	---	0.022	0.030
Maintenance (fully burdened)	---	---	---	---	---	0.071	0.074
Max envelope length	80	64	7,586	11	3	0.010	0.014
Max envelope height	110	91	10,743	16	4	0.015	0.020

Baseline Characteristics

TOGW = 41,730 lbm
Mfg empty weight = 26,076 lbm
Flyaway cost = $4,835,535

Block fuel (600 nm) = 3231 lbm
Block fuel (100 nm) = 813 lbm

DOC (600 nm) = 5.383 ¢/seat nm
DOC (100 nm) = 7.356 ¢/seat nm



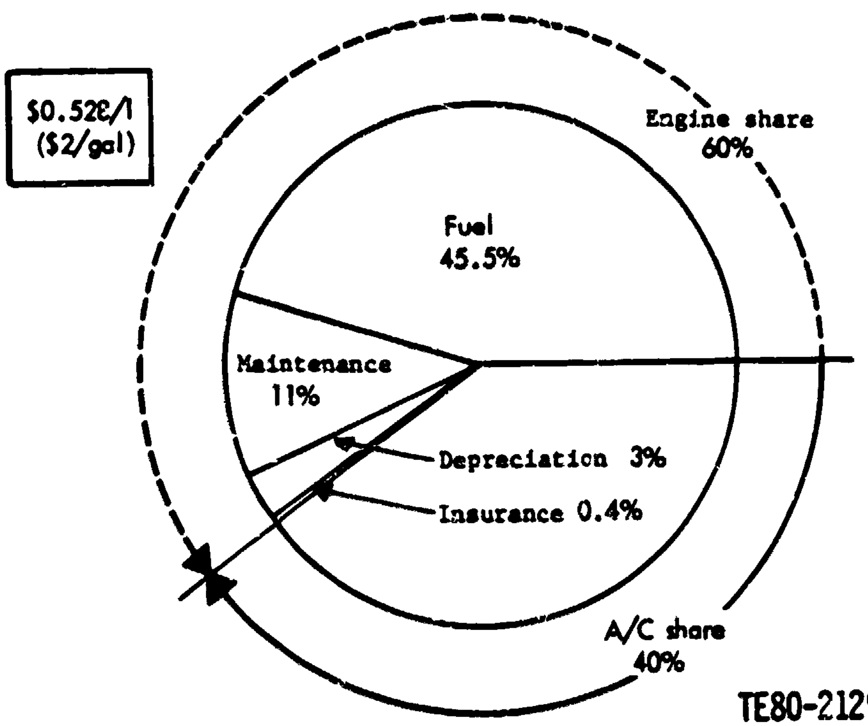


Figure 17. - Typical DOC breakdown--high speed aircraft.

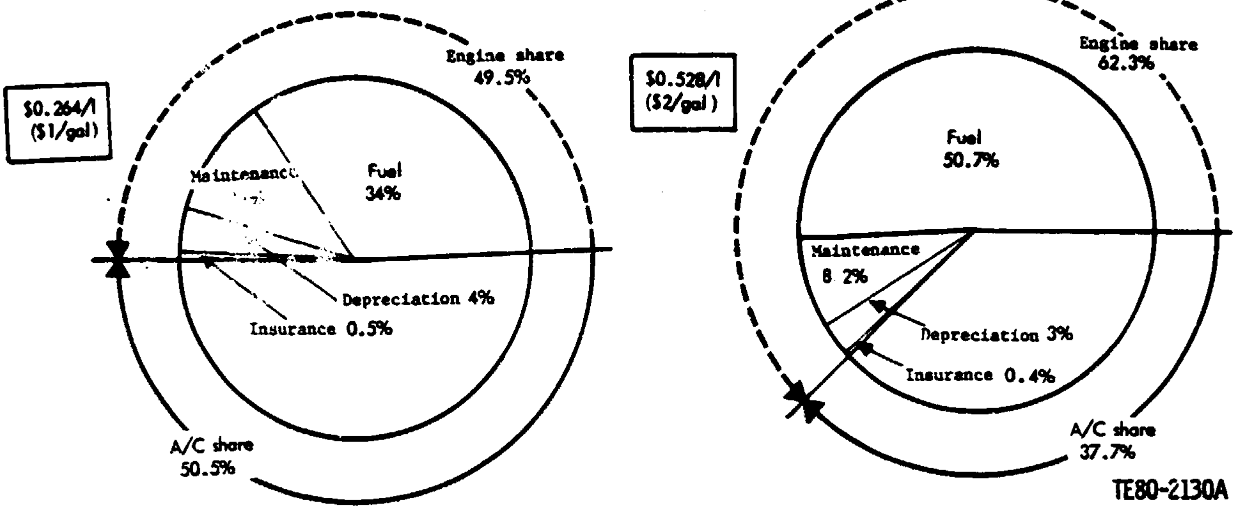


Figure 18. - Typical DOC breakdown--low speed aircraft.

TABLE XV. - PROPELLER SENSITIVITY DATA

Propeller parameter improvement	Reduction in DOC--% (DDA 0.7 Mach aircraft)
1% Δ propeller efficiency	0.350
1% Δ propeller weight (4.22 kg (9.3 lbm)/propeller)	0.024
1% Δ propeller OEM price ($509/propeller)	0.006

ADVANCED TECHNOLOGY IDENTIFICATION AND EVALUATION

The objectives of this portion of the STAT program were to conduct the following analyses as applicable to the turbine engine size requirements for 30- and 50-passenger commuter aircraft:

- o Conduct parametric engine/airframe studies to optimize cycle and design arrangement
- o Identify technology advances
- o Screen and select those with best payoff potential
- o Define and describe candidate advanced technology engines
- o Determine the payoff potential by comparing the advanced technology engine with both the current technology engine and its 1985 time frame derivative

CYCLE SELECTION

Airframe sensitivities to engine parameters were developed in conjunction with vehicle specialists as explained under Baseline Airplane and Missions. The airframe sensitivities, or airframe/mission partial derivatives as they are denoted in Figure 19, are obtained from the airframe mission model and applied in combination with engine performance, weight, and cost partial derivatives from the engine cycle/performance model, engine weight model, and engine cost model to obtain a net variation in the payoff parameter. Figure 21 shows this process in detail, using a compressor advanced technology element evaluated in terms of a net change in DOC for the 185.2 km (100 nm) route segment.

Sensitivities to changes in the baseline PD370-37 engine cycle were determined using major segments of the design 1111.2 km (100 nm) mission for the high speed airplane. These major segments were determined from the DDA mission analysis model. The results from this model correlated closely with those of Lockheed. Table XVI shows the total fuel burned in the mission and the percent of the total burned in each mission segment. Climb, cruise, and loiter use the major portion of the fuel. Power, speed, altitude, and power setting are defined for these major segments at the bottom of Table XVI. Since climb and cruise/loiter mean operating conditions for both the design and alternate missions were almost identical, the six operating conditions were reduced to four for the sensitivity analysis.

To develop the sensitivities to changes in the baseline Model PD370-37 engine cycle, the parameters shown in Table XVII were changed individually at each of the four mission operating conditions shown in Table XVIII. The baseline engine performance was obtained by running the mission operating conditions at the BOT's shown in Table XVIII.

The effects of these engine cycle parameter variations on engine performance for one of the four operating conditions are shown in Tables XIX and XX. The representative mission fuel used (approximately 81% to 89%) was determined by using the mission segment times shown in Table XVI. The sensitivity of 185.2- and 1111.2 km (100- and 600 nm) mission fuel used to the changes in engine cycle parameters is shown in Tables XXI and XXII. The change in fuel used, from that of the baseline engine, expressed in percent, is the mission weighted percent change in sfc from those of the baseline PD370-37 engine. These sensitivities are shown graphically in Figures 20 through 22.

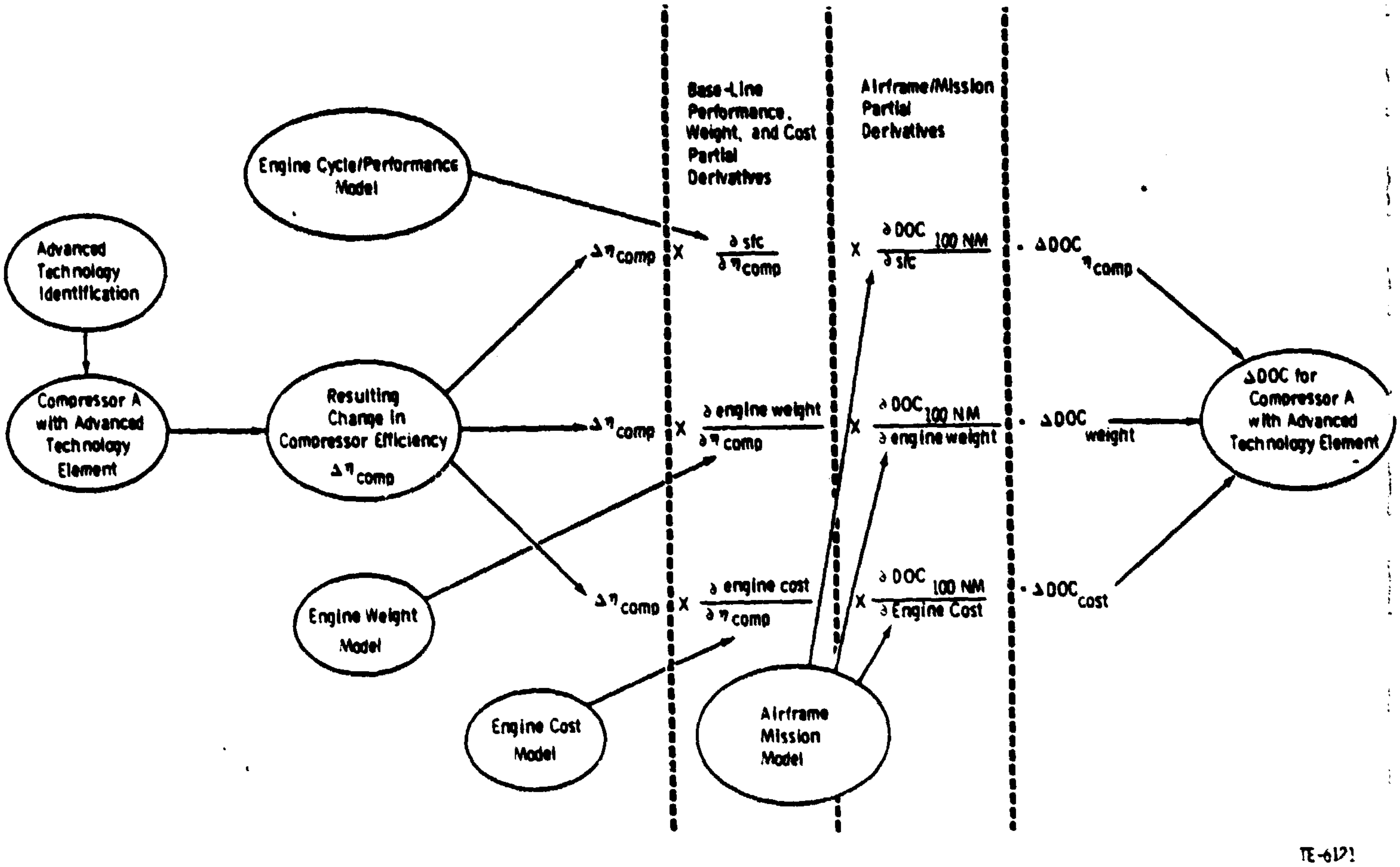


Figure 19. - Methodology using sensitivity data.

TABLE XVI. - DDA MISSION ANALYSIS RESULTS--CURRENT TECHNOLOGY HIGH SPEED AIRCRAFT

	Design mission 1111.2 km (600 nm)			Alternate mission 185.2 km (100 nm)		
Mission phases	Time--min	Fuel--kg (lbm)	Fuel--%	Time--min	Fuel--kg (lbm)	Fuel--%
Take-off	1.0	- -	1.9	1.0	- -	3.1
Climb	15.6	- -	15.7	5.1	- -	12.4
Cruise	77.8	- -	43.3	10.5	- -	17.2
Fuel Reserves						
Take-off	1.0	- -	1.9	1.0	- -	3.1
Climb	2.1	- -	3.5	2.3	- -	6.2
Cruise	20.7	- -	11.2	20.0	- -	18.8
Loiter	45.0	- -	22.5	45.0	- -	39.2
Totals	163.2	1932.7 (4260.8)	100.0	84.9	1156.4 (2549.5)	100.0

Note: Climb--6,096 m (20,000 ft)/0.5 M_N/climb power--both missions
Cruise--10,668 m (35,000 ft)/0.7 M_N/86% max cont--design mission
6,096 m (20,000 ft)/0.7 M_N/83% max cont--alternate mission
Cruise/loiter--3,048 m (10,000 ft)/0.4 M_N/27% max cont--both missions

TABLE XVII. - CYCLE PARAMETER CHANGES

Cycle parameter	Change
Compressor efficiency	± 3% and -6%
HP turbine efficiency	± 3%
LP turbine efficiency	± 3%
Turbine cooling air	± 25%
Burner pressure drop	± 25%

TABLE XVIII. - MISSION OPERATING CONDITIONS

Condition	Altitude--m (ft)	Mach No.	Baseline engine BOT--K (°R)	Baseline engine power--kW (shp)
Cruise/loiter	3,048 (10,000)	0.4	1036 (1865)	1456 (1953)
Climb	6,096 (20,000)	0.5	1480 (2664)	3811 (5110)
Cruise	6,096 (20,000)	0.7	1343 (2418)	3747 (5025)
Cruise	10,668 (35,000)	0.7	1311 (2360)	2174 (2915)

TABLE XIX. - PD370-37 SENSITIVITY STUDY OF CYCLE CHARACTERISTICS AT 6096 m CRUISE (SI UNITS)

(altitude: 6096 m; standard day; velocity: 0.7 M_N; power: cruise)

	Compressor η			HP turbine η		LP turbine η		Turbine cooling		Burner ΔP		
Sensitivity change--%	+3	-3	-6	+3	-3	+3	-3	+25	-25	+25	-25	Cumulative*
$N/\sqrt{\theta}$ compressor--%	99.9	100.0	100.1	99.9	99.9	99.9	99.9	99.9	99.9	100.0	99.9	99.9
W_{in}--kg/s	12.12	12.87	13.35	12.22	12.75	12.10	12.85	12.74	12.21	12.54	12.40	11.30
$W\sqrt{\theta}/\delta$--kg/s	18.50	19.65	20.38	18.65	19.46	18.47	19.62	19.45	18.64	19.14	18.93	17.25
R_{comp}	11.48	11.50	11.52	11.48	11.49	11.49	11.49	11.49	11.49	1.49	11.49	11.47
η_{comp}--%	86.35	81.29	78.75	83.83	83.81	83.82	83.82	83.82	83.83	83.82	83.83	86.35
$\Delta P/P_{burn}$--%	4.13	4.14	4.14	4.13	4.14	4.13	4.13	4.14	4.14	5.17	3.10	3.10
RIT--K	1312.7	1312.5	1311.4	1312.9	1311.4	1312.7	1312.8	1311.7	1313.3	1312.6	1312.6	1312.2
$N/\sqrt{\theta}$ HPT--%	99.4	99.5	99.6	99.4	99.5	99.4	99.4	99.4	99.4	99.4	99.4	99.4
η_{HPT}--%	88.18	88.19	88.20	90.83	85.54	88.19	88.19	88.19	88.19	88.19	88.18	90.83
Cooling HP turbine--% HPC inlet air	4.0	4.0	4.0	4.0	4.0	4.0	4.0	5.0	3.0	4.0	4.0	3.0
$N/\sqrt{\theta}$ LPT--%	101.9	101.9	102.0	101.9	101.9	101.9	101.9	101.9	101.9	101.9	101.9	101.9
η_{LPT}--%	90.01	90.97	91.46	90.01	90.97	93.19	87.77	90.70	90.26	90.61	90.35	91.94
Cooling LP turbine--% HPC inlet air	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.25	0.75	1.0	1.0	0.75
SFC--mg/W·s	62.89	65.09	66.45	62.67	65.27	62.01	65.88	64.33	63.48	64.24	63.55	59.37

Note: All runs made to constant (base value) shaft power.
*+3% η_C, +3% η_{HPT}, +3% η_{LPT}, -25% T_C, -25% burner ΔP.

TABLE XX. - PD370-37 SENSITIVITY STUDY OF CYCLE CHARACTERISTICS AT 20,000 ft CRUISE (CUSTOMARY UNITS)

(altitude: 20,000 ft; standard day; velocity: 0.7 M_N; power: cruise)

Sensitivity change--%	Compressor η			HP turbine η		LP turbine η		Turbine cooling		Burner ΔP		Cumulative*
	+3	-3	-6	+3	-3	+3	-3	+25	-25	+25	-25	
$N/\sqrt{\theta}$ compressor--%	99.9	100.0	100.1	99.9	99.9	99.9	99.9	99.9	99.9	100.0	99.9	99.9
W_{in}--lb/sec	26.72	28.38	29.43	26.94	28.12	26.68	28.34	28.08	26.92	27.64	27.33	24.91
$W\sqrt{\theta}/\delta$--lb/sec	40.79	43.32	44.92	41.13	42.92	40.72	43.25	42.86	41.10	42.19	41.72	38.02
R_{comp}	11.48	11.50	11.52	11.48	11.49	11.49	11.49	11.49	11.49	11.49	11.49	11.47
η_{comp}--%	86.35	81.29	78.75	83.83	83.81	83.82	83.82	83.82	83.83	83.82	83.83	86.35
$\Delta P/P_{burn}$--%	4.13	4.14	4.14	4.13	4.14	4.13	4.13	4.14	4.14	5.17	3.10	3.10
RIT--R	2362.8	2362.5	2360.6	2363.3	2360.6	2362.8	2363.0	2361.1	2363.9	2362.7	2363.0	2362.0
$N/\sqrt{\theta}_{HPT}$--%	99.4	99.5	99.6	99.4	99.5	99.4	99.4	99.4	99.4	99.4	99.4	99.4
η_{HPT}--%	88.18	88.19	88.20	90.83	85.54	88.19	88.19	88.19	88.19	88.19	88.18	90.83
Cooling HP turbine--% HPC inlet air	4.0	4.0	4.0	4.0	4.0	4.0	4.0	5.0	3.0	4.0	4.0	3.0
$N/\sqrt{\theta}_{LPT}$--%	101.9	101.9	102.0	101.9	101.9	101.9	101.9	101.9	101.9	101.9	101.9	101.9
η_{LPT}--%	90.01	90.97	91.46	90.01	90.97	93.19	87.77	90.70	90.26	90.61	90.35	91.94
Cooling LP turbine--% HPC inlet air	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.25	0.75	1.0	1.0	0.75
SFC--lbm/hr·HP	0.3722	0.3852	0.3933	0.3709	0.3863	0.3670	0.3899	0.3306	0.3757	0.3802	0.3761	0.3514

Note: All runs made to constant (base value) shaft power.
*+3% η_C, +3% η_{HPT}, +3% η_{LPT}, -25% T_C, -25% burner ΔP.

TABLE XXI. - PD370-37 SENSITIVITY STUDY OF MISSION FUEL CHANGES (SI UNITS)

Sensitivity change--%	Baseline	Compressor η			HP turbine η		LP turbine η		Turbine cooling		Burner ΔP		Cumulative*
		+3	-3	-6	+3	-3	+3	-3	+25	-25	+25	-25	
185.2-km mission--fuel used--kg													
Cruise/loiter	518.4	500.2	540.0	564.6	497.8	542.5	503.7	534.6	527.8	510.5	525.3	512.5	455.6
Climb	148.3	145.9	151.2	154.4	145.5	151.6	143.9	152.9	149.3	147.3	149.1	147.6	137.9
Cruise	150.7	148.5	153.8	156.9	148.1	154.1	146.5	155.6	151.9	150.0	151.8	150.1	140.2
Total fuel--kg	817.4	794.6	844.9	876.0	791.3	848.2	794.1	843.1	829.0	807.8	826.1	810.2	733.7
Change in fuel--%		-2.78	3.37	7.16	-3.19	3.76	-2.86	3.14	1.41	-1.18	1.06	-0.89	-10.24
1111.2-km mission--fuel used--kg													
Cruise/loiter	524.0	505.6	545.9	570.7	503.1	548.3	509.1	540.4	533.4	516.0	530.9	518.0	460.6
Climb	310.6	305.7	316.7	323.6	304.8	317.6	301.5	320.3	312.8	308.7	312.4	309.1	289.0
Cruise	644.6	1035.8	654.6	667.0	634.0	656.4	625.8	664.6	648.2	641.1	648.2	641.7	602.3
Total fuel--kg	1479.2	1447.1	1517.2	1561.2	1442.0	1522.4	1436.5	1525.4	1494.4	1465.9	1468.9	1468.9	1324.7
Change in fuel--%		-2.17	2.57	5.54	-2.52	2.92	-2.89	3.12	1.03	-0.90	0.83	-0.70	-8.61

*+3% η_C, +3% η_{HPT}, +3% η_{LPT}, -25% T_C, -25% burner ΔP.

TABLE XXII. - PD370-37 SENSITIVITY STUDY OF MISSION FUEL CHANGES (CUSTOMARY UNITS)

Sensitivity change--%	Baseline	Compressor η			HP turbine η		LP turbine η		Turbine cooling		Burner ΔP		Cumulative*
		+3	-3	-6	+3	-3	+3	-3	+25	-25	+25	-25	
100-NM mission--fuel used--lbm													
Cruise/loiter	1142.9	1102.8	1190.5	1244.7	1097.4	1196.0	1110.4	1178.6	1163.5	1125.5	1158.0	1129.9	1004.5
Climb	326.9	321.7	333.3	340.5	320.8	334.2	317.3	337.1	329.2	324.8	328.7	325.3	304.1
Cruise	332.3	327.4	339.0	346.0	326.4	339.7	322.9	343.0	334.9	330.6	334.6	330.9	309.0
Total fuel--lbm	1802.1	1751.9	1862.8	1931.2	1744.6	1869.9	1750.6	1858.7	1827.6	1780.9	1821.3	1786.1	1617.6
Change in fuel--%		-2.78	3.37	7.16	-3.19	3.76	-2.86	3.14	1.41	-1.18	1.06	-0.89	-10.24
600-NM mission--fuel used--lbm													
Cruise/loiter	1155.2	1114.7	1203.4	1258.1	1109.2	1208.9	1122.4	1191.4	1176.0	1137.7	1170.5	1142.1	1015.4
Climb	684.8	674.0	698.3	713.4	672.0	700.3	664.8	706.2	689.7	680.6	688.8	681.5	637.2
Cruise	1421.2	1401.7	1443.2	1470.4	1397.8	1447.1	1379.7	1465.3	1429.0	1413.4	1429.0	1414.7	1327.8
Total fuel--lbm	3261.2	3190.4	3344.9	3441.9	3179.0	3356.3	3166.9	3362.9	3294.7	3231.7	3288.3	3238.3	2920.4
Change in fuel--%		-2.17	2.57	5.54	-2.52	2.92	-2.89	3.12	1.03	-0.90	0.83	-0.70	-8.61

*+3% η_C, +3% η_{HPT}, +3% η_{LPT}, -25% T_C, -25% burner ΔP.

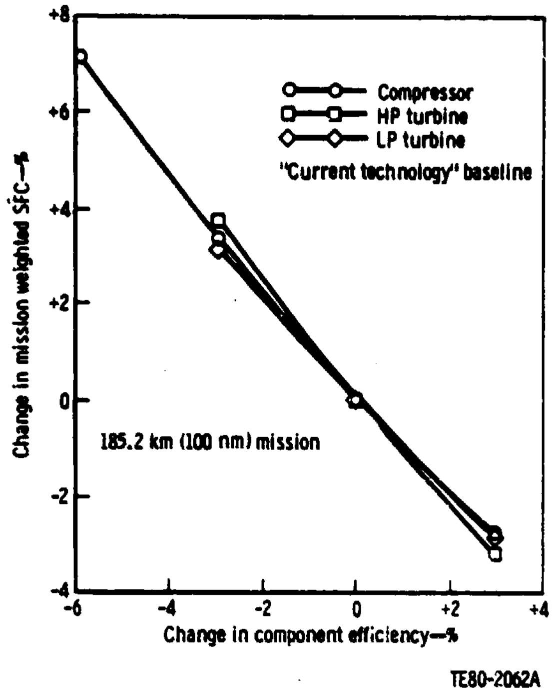


Figure 20. - Engine sfc sensitivity to component efficiency—high speed airplane, alternate mission.

Sensitivities to changes in the baseline Model PD370-37 engine cycle were also determined using major segments of the design 1111.2 km (600 nm) mission and of the alternate 185.2 km (100 nm) mission, for the low speed airplane. These sensitivities were determined in a similar manner to the methods described for the high speed airplane. It was found that sensitivities to component efficiency, cooling air, and burner pressure drop were essentially the same for the low and high speed airplane missions.

A matrix of engines was postulated at the nominal STAT engine sizes of 1790 and 3579 kW (2400 and 4800 shp). Baseline compressor, high pressure turbine, and low pressure turbine efficiencies and cooling air amounts were assumed as indicated in Tables XXIII and XXIV for each engine in the matrix. The matrix included compressor pressure ratios of 5, 10, 15, and 25 over a range of turbine rotor inlet temperatures (RIT) as shown.

Compressor efficiencies were specified for the STAT nominal size engines with consideration for clearance effects using axial staging arrangements.

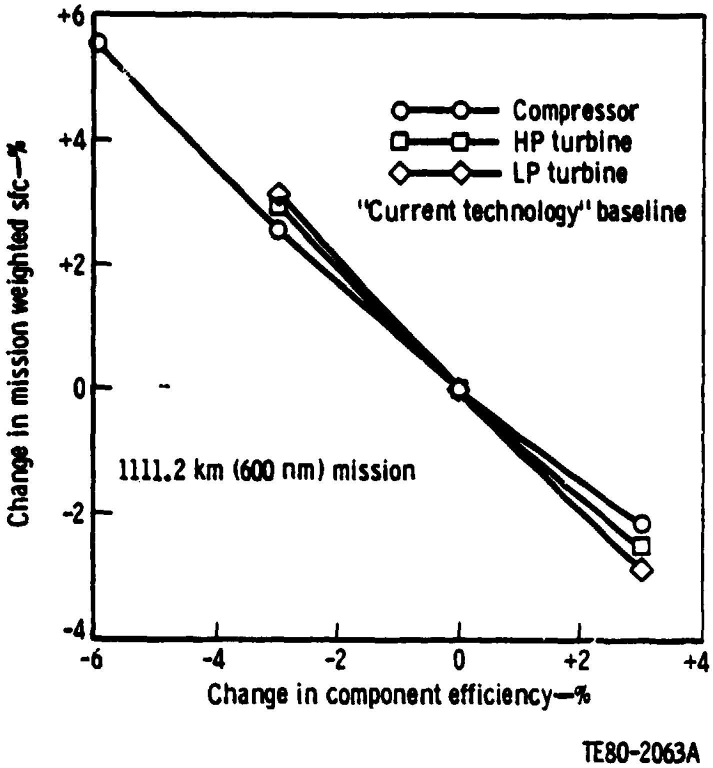


Figure 21. - Engine sfc sensitivity to component efficiency--high speed airplane, design mission.

TABLE XXIII. - STAT SENSITIVITY STUDY--CYCLE PARAMETERS FOR 1790 kW (2400 shp) ENGINE

Compressor η (polytropic) 88.0%

		RIT--K (°F)			
	R_c	1506 (2250)	1644 (2500)	1783 (2750)	1950 (3050)
HP turbine η (adiabatic)--%	5	88.8	88.6	88.4	88.3
	10	87.5	87.2	86.9	86.7
	15	86.4	86.1	85.7	85.5
	25	85.3	84.8	84.5	84.1
LP turbine η (adaibatic)--%	5	90.1	89.9	89.8	89.7
	10	89.3	89.0	88.7	88.4
	15	89.0	88.6	88.2	87.8
	25	89.1	88.5	88.0	87.6
Cooling air--%	5	2.67	6.57	9.52	14.1
	10	3.38	4.53	8.8	13.7
	15	6.40	8.80	13.9	16.1
	25	12.0	16.7	25.6	27.7

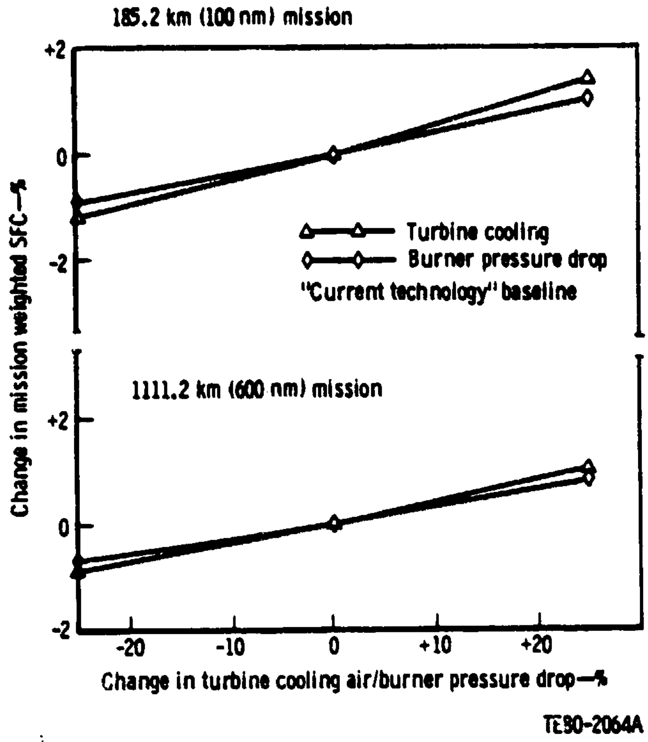


Figure 22. - Engine sfc sensitivity to turbine cooling air and burner pressure drop--high speed airplane.

TABLE XXIV. - STAT SENSITIVITY STUDY-CYCLE PARAMETERS FOR 3579 kW (4800 shp) ENGINE

Compressor η (polytropic) 89.0%

	R_c	RIT--K (°F) 1506 (2250)	1644 (2500)	1783 (2750)	1950 (3050)
HP turbine η	5	92.1	92.0	91.9	91.8
(adiabatic)--%	10	90.8	90.6	90.4	90.2
	15	90.0	89.7	89.4	89.1
	25	89.0	88.4	88.1	87.7
LP turbine η	5	91.8	91.7	91.6	91.5
(adaibatic)--%	10	91.8	91.1	90.9	90.8
	15	91.2	90.8	90.6	90.3
	25	91.2	90.8	90.5	90.1

TABLE XXIV. (CONT)

	R_c	RIT--K (°F) 1506 (2250)	1644 (2500)	1783 (2750)	1950 (3050)
Cooling air--%	5	2.68	6.57	9.52	14.1
	10	3.38	4.53	8.8	13.7
	15	6.40	8.80	13.9	16.1
	25	12.0	16.7	25.6	27.7

HP and LP turbine efficiencies and cooling flow rates were projected to reflect the effects of engine flow rate, cycle pressure ratio, and turbine RIT upon the turbine performance. The coolant flow rates and efficiency levels were the result of the changing turbine physical size, work requirements, and number of stages. Coolant flows were established consistent with common life requirements based upon cycle temperature, cooling air termperature, stage work requirement, and number of stages that require cooling. The associated cooling effectiveness was chosen with a consideration given to the significant impact of initial engine cost on DOC. Exotic cooling schemes and their attendant high engine costs were considered only at the highest temperature levels where they are an absolute necessity.

The effect of these cycle considerations on relative mission weighted-sfc is shown in Figures 23 and 24. The results tend to localize the region of interest for both engine sizes at 15 to 25 pressure ratio at RIT from 1506 to 1644 K (2250 to 2500°F). These results were determined by adjusting design point sfc's calculated for the engine matrix for changes in compressor and turbine efficiency and cooling air quantities from the nominal values used in generation of the sensitivity data to the values shown in Tables XXIII and XXIV. The mission fuel weighted sensitivities were used as previously described. Minimums occured where improved cycle efficiency due to increase in pressure ratio and turbine temperature was overcome by reduced component adiabatic efficiency and performance penalties due to increased cooling air.

Further study indicated that an RIT of 1506 K (2250°F) was an acceptable development risk for the 1988 time period. The risk attendant with higher RIT was judged excessive for the small improvement in sfc realized.

At a turbine temperature of 1506 K (2250°F), further parametric studies were made to determine the effect of compressor pressure ratio condsidering engine weight, cost, and dimensions in addition to mission-weighted sfc. Table XXV presents these results for the 3579 kW (4800 shp) engine in terms of percent change in weight, cost, length, diameter, and sfc from a 4800 shp reference engine. Using the sensitivity data developed for the high speed commuter airplane, engine characteristic changes were converted to a percent change in DOC at 185.2 km (100 nm) and plotted in Figure 25. These results indicate that minimum DOC is obtained at a pressure ratio of approximately 20:1.

Summarizing the results of the cycle analisis, an RIT of 1506 K (2250°F), and a compressor pressure ratio of 20:1 were judged reasonable selections for both the 1790- and 3579 kW (2400- and 4800 hp) engines.

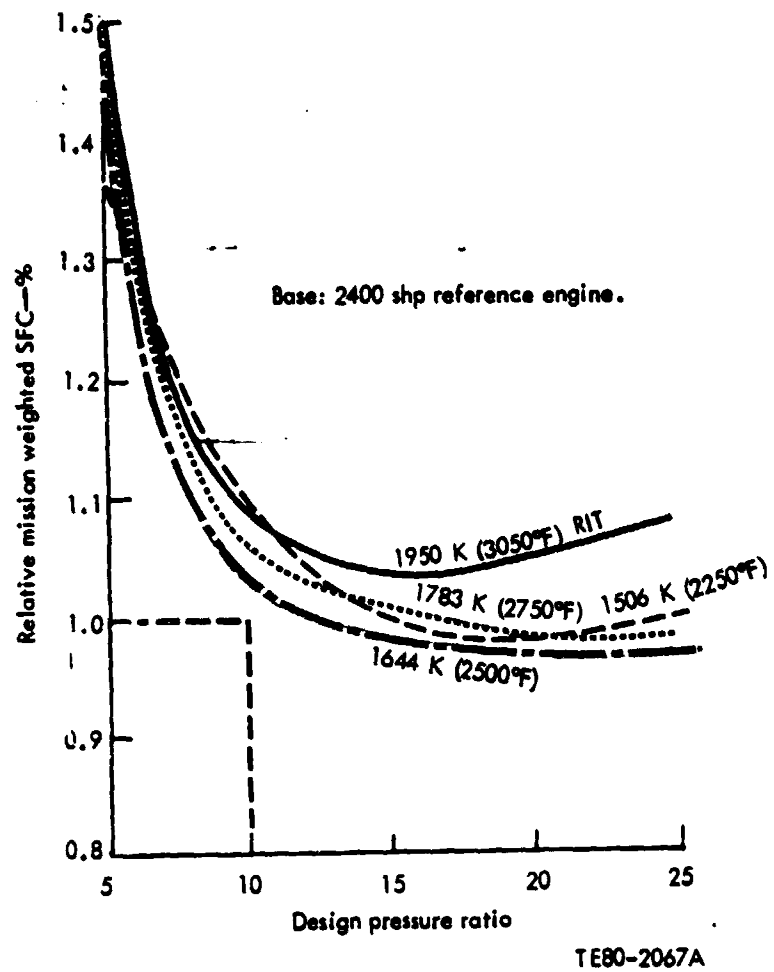


Figure 23. - Mission weighted sfc trends--1790 kW (2400 shp) size.

TABLE XXV. - STAT ENGINE PRESSURE RATIO SELECTION 3579 kW (4800 shp) SLSS

	1	2	3	4
R_c	5	10	15	25
RIT--K (°F)	1506 (2250)	1506 (2250)	1506 (2250)	1506 (2250)
% Δ weight	-10.0	-21.9	-23.8	-21.2
% Δ cost	-8.3	-32.1	-20.5	-18.11
% Δ length	+3.98	-8.5	-11.0	-10.4
% Δ diameter	+14.7	+2.4	+0.0005	+2.1
% Δ sfc	+29.0	+2.0	-8.0	-7.1
% Δ DOC (100 nm)	+9.49	-4.24	-6.467	-5.66

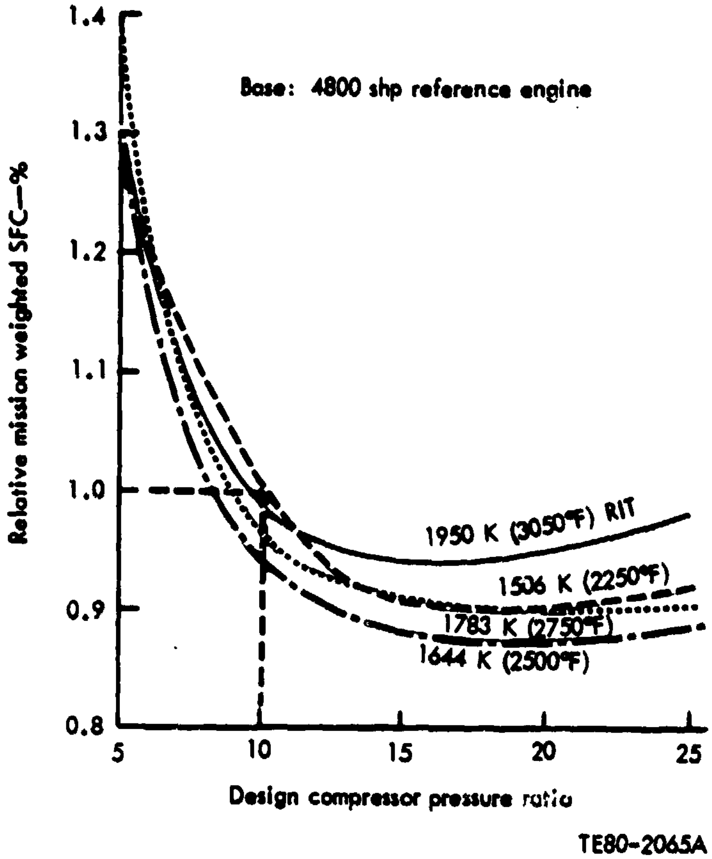


Figure 24. - Mission weighted sfc trends--3579 kW (4800 shp) size.

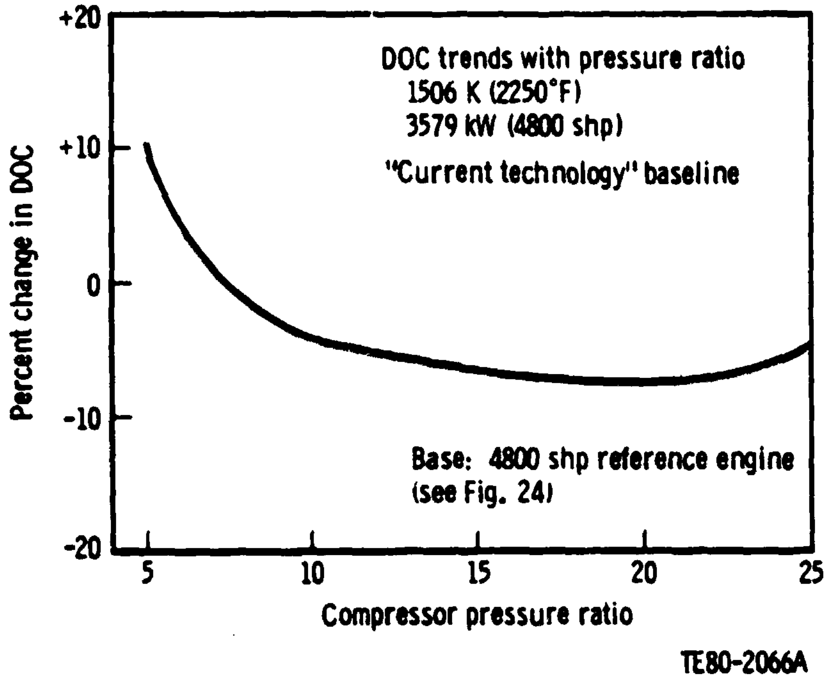


Figure 25. - DOC trends with pressure ratio at 1506 K (2250°F) and 3579 kW (4800 shp) conditions.

CONFIGURATION TRADES

An extensive list of advanced technology candidates for incorporation in the STAT engines was prepared early in Task II. These candidates were screened and correlated in four major areas:

- o Rotating components
- o Static components
- o Propulsors and drives
- o Nacelles and accessories

Potential technology items were first screened to determine those for which sufficient technical information and background experience existed at DDA to conduct a trade study. Another criterion considered in this initial screening was the extent of effort required to obtain baseline data upon which to make tradeoff assessments. In some cases, the effort required was judged to be beyond the scope of this study program. Advanced technologies associated with propellers were furnished to DDA by NASA.

After this initial screening, individual components in each area were investigated to determine the applicability of the technology and the estimated change in component characteristics (efficiency, weight, cost, life, etc). At the same time, the associated risk level was noted to aid in the screening of those technology elements with the greatest promise. Emerging advanced technology areas were investigated to determine their potential impact on powerplants sized to meet the STAT commuter aircraft requirements, particularly in areas that were unique such as the shorter duration flight cycle, and emphasis on reduced initial costs and operating costs.

The study engine characteristics were also evaluated in terms of advanced design features that might be incorporated into the STAT engines. The same screening process and study limitation criteria that were used in selecting technology trade study candidates were applied to determine the design features to be evaluated. Each design feature selected was studied with respect to the measureable improvements in components, cycle, or engine compared to the STAT baseline engines. As with the technology elements, the associated risks were judged and entered into the evaluation.

The baseline engine sensitivity parameters were applied to technology item and design features to generate the resulting partial derivatives of engine performance/cost parameters (sfc, initial cost, weight, etc). Those judged to have the most merit were subjected to the airframe sensitivity analysis. Each selected design change (in terms of engine performance parameters) was then checked to determine the resized airframe (DOC), empty mass, acquisition cost, fuel consumed, etc, for the baseline mission. Those design features with the greatest merit were chosen for further evaluation for the STAT engines.

The lists of technology items and design features that were finally evaluated, are shown in Table XXVI. This table indicates the recommendation applicable to each item indicating whether it was selected for incorporation in the STAT engines or was rejected. The table also shows whether this decision was based on a DOC evaluation or was one based on judgment since the DOC impact could not be ascertained.

TABLE XXVI. - CANDIDATE ADVANCED TECHNOLOGIES AND DESIGN FEATURES

CANDIDATE ADVANCED TECHNOLOGIES	Selected		Rejected	
Technology Item	DOC	Judgement	DOC	Judgement
Power Section				
o Compressor				
Cycle pressure ratio	X			
Configuration (axial & axial/cent.)		X		
Hybrid centrifugal impeller	X			
High temperature titanium aft wheels		X		
Welded titanium spool	X			
o Turbine				
Rotor inlet temperature	X			
Configuration -				
Axial		X		
Radial inflow				X
Hybrid rotors, composite shafts & supercritical shafts	X			
Thermal barrier coatings			X	
Ceramic blades & vanes and airfoil coatings				X
Cast-in impingement cooling	X			
Long-life bearings	X			
o Combustor/diffusers				
Transpiration cooled combustor	X			
Vortex controlled diffuser	X			
o Engine accessories				
Electronic fuel control	X			
Fuel pump and metering system	X			
Engine condition monitoring	X			
o Noise reduction				
Compressor design	X			
Combustor design	X			
Reduction Gear				
o Advanced design				
Composite materials	X			
Steel/titanium gears				X
Finite element gear analysis		X		
Superplastic formed titanium				X

TABLE XXVI (CONT)

CANDIDATE ADVANCED DESIGN FEATURES

Design Feature	Selected DOC	Selected Judgement	Rejected DOC	Rejected Judgement
Power Section				
o Compressor				
Erosion resistant design	X			
Inlet particle seperator	X			
Rotor/case response to rotating stall	X			
Clearance control -				
Active				X
Straddle mounts	X			
Thermally matched rotor/case	X			
o Turbine				
Clearance Control -				
Active	X			
Straddle mounts			X	
Thermally matched rotor/case	X			
Leakage control		X		
Abradable coatings		X		
Advance Bearings -				
Beryllium-backed bearing races				X
Tapered bearings				X
Other				
Modular construction		X		
Remote accessories		X		
Advanced propellers	X			
Turbofan engine				X
Power extraction, mechanical vs air bleed		X		
Reduced weight nacelles		X		

The following paragraph describes the analyses by engine section. The impacts on engine weight, performance, maintainability, and cost are given where tradeoffs could be made; otherwise decisions were based on judgement.

Compressors

Compressor configurations for both engines were determined and are shown in Figures 26 and 27 for the 1790- and 3579 kW (2400- and 4800 hp) engines, respectively. The compressor pressure ratio selected was 20:1, as previously discussed. In selecting these configurations, the experience derived from recent compressor studies, including the small compressor study for NASA/LeRC, was employed. For tradeoff examination of alternate compressor arrangements, trends were studied that show adiabatic compressor efficiency versus design flow for axial, axial-centrifugal, and one- and two-spool, two-stage centrifugal compressors. The compressors were selected to be axial-centrifugal and axial, for the 1790- and 3579 kW (2400- and 4800 shp) engines, respectively. Preliminary compressor aerothermodynamic design data showed axial compressors

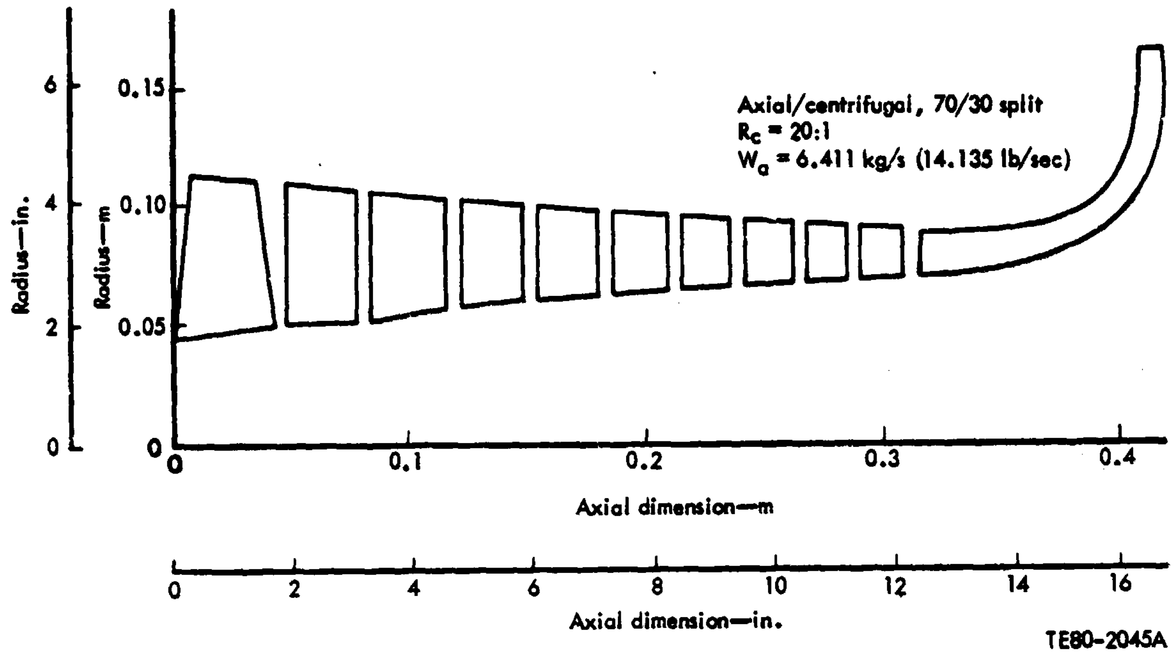


Figure 26. - STAT advanced compressor flow path--1790 kW (2400 hp) engine.

at the study pressure ratio have higher design efficiencies than other configurations over the study range of flow rates. An axial compressor was configured for both engines initially. The small compressor, however, was found to have extremely small airfoils at the aft end. These small airfoils would be difficult to manufacture as well as to make the compressor efficiency very sensitive to small changes in clearance. For this reason an axial-centrifugal compressor was selected for the small engine. An axial compressor was selected for the large engine.

Hybrid Compressor Impeller Rotor

An evaluation was made of the hybrid centrifugal compressor impeller rotor. This item uses hot isostatic press (HIP) bonding to attach a cast rim with blades to a forged bore insert.

A recent value engineering study of the Model 250-C30 impeller rotor was made in which a similar substitution was considered. This analysis showed that the cost reduction potential was 71%. Using this ratio and estimated cost of the 1790 kW (2400 shp) STAT engine compressor impeller, the resulting change in engine cost would be -2.3% with a corresponding change in DOC of -0.25%.

This item was selected for the STAT 1790 kW (2400 shp) STAT engine.

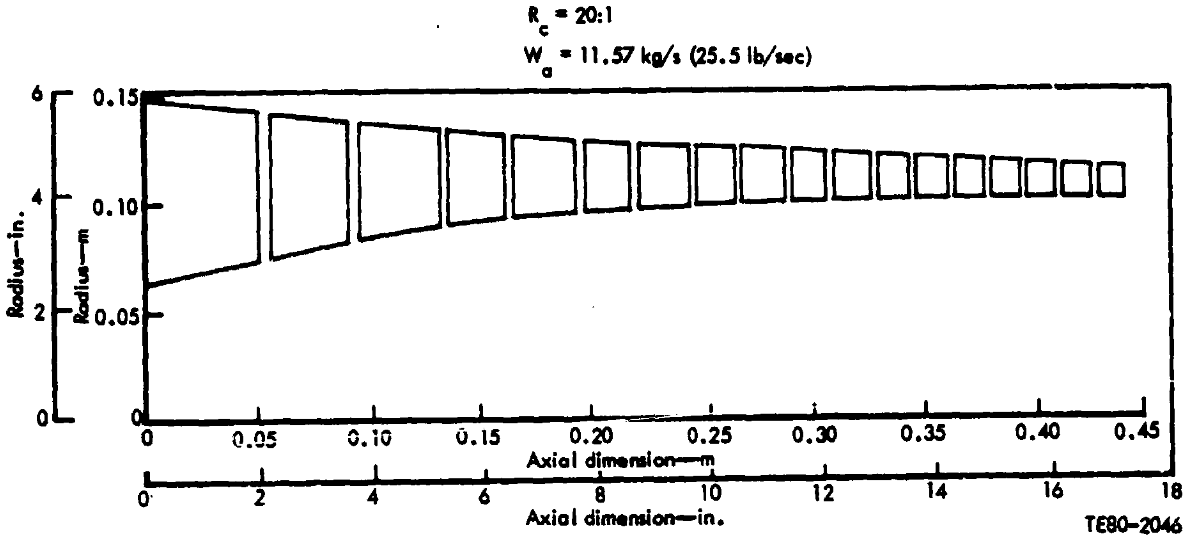


Figure 27. - STAT advanced compressor flow path--3579 kW (4800 hp) engine.

High Temperature Aft Titanium Wheels

In both STAT engine sizes the cycles selected have compressor discharge temperatures, at rated conditions in the range of 738.7 to 744.3 K (870° to 880°F). At these temperatures, alloys such as 6-4 Titanium have little creep strength. These alloys are thus inappropriate for wheels to be used in the latter stages of the compressors.

IMI829, a high temperature Titanium alloy developed outside the U.S., appears to be applicable to this aft wheel location, based on preliminary data. However, data on alloy weldability is not available, and this may impact its selection for a welded drum rotor construction. The application of IMI829 to the 2400 shp compressor is uncertain at this time due to the hybrid construction of its centrifugal impeller.

Welded Titanium Spools

Advanced compressors in development today use welded drums to provide greater stability and reduced weight. Improvements can be made in design and fabrication technology to further reduce weight and cost by simplifying the configuration to facilitate welding in remote or blind areas. Based on experience with other DDA turboshaft engines, it is estimated that the following savings may be achieved with the STAT engines:

Rated power -- kW (shp)	% Δ Weight	% Δ Cost	% Δ DOC
1790 (2400)	-1.0	-0.6	-0.093
3579 (4800)	-1.8	-1.1	-0.196

Erosion Protection

DDA experience with T56 engines operating in desert climates has shown that severe sand and dust erosion damage to an axial compressor can cause the engine to be removed for overhaul in just a few hundred hours.

The STAT engines will be designed to be more tolerant of injested dirt particles in the airstream than current technology engines. If the STAT engines operate in an extremely dusty environment, their normal MTBR may be reduced by one-half. The addition of suitable inlet particle separators (IPS) would restore their normal MTBR, and would provide the following inpact to the cost and operation of the engines:

Rated power--kW (hp)	% Δ Weight	% Δ Cost	% Δ sfc	% Δ Maint. Cost	% Δ DOC
1790 (2400)	+5.0	+2.5	+1.0	-14.4	-0.818
3579 (4800)	+3.0	+2.0	+1.0	-10.4	-0.633

Although the IPS adds to the engine weight, cost and fuel usage, the gain in lowered maintenance cost could more than offset this disadvantage.

Prediction of Rotor/Case Response to Rotating Stall

The small diameter, highly loaded compressors of the STAT engines require close blade tip clearances to achieve the desired performance. It is essential that the design of these advanced compressors consider the dynamic behavior of the rotors and cases during surge, rotation stall, and rapid thermal changes. Without the ability to predict these phenomena, it would be necessary to design with greater tip clearance and to provide either an additional compressor stage or an additional bearing and attendant support structure. With the ability to predict the rotor/case response to rotating stall, however, these penalties could be removed with the following typical improvement to DOC:

Rated power--kW (hp)	% Δ Weight	% Δ Cost	% Δ sfc	% Δ DOC
1790 (2400)	-5.0	-8.0	-0.8	-1.334
3579 (4800)	-6.0	-6.0	-0.8	-1.234

Clearance Control

The payoff per stage with compressor active clearance control is such that stage efficiency improves 3% for each 1% of blade height change in tip clearance. The highest payoff occurs in the aft compressor stages where blade height is smallest. This complicates the design, particularly where a vane actuating system is desirable from an acceleration/surge margin point of view.

Simple mechanical systems are essentially eliminated if a vane actuation system is already employed. Thermal systems probably would not pay off since the "on" system time is short and because of the penalty to the cycle. Active clearance control was, therefore, rejected for the STAT engines. Straddle rotor mounts and thermally matched rotors, blades, vanes, and the case were incorporated in the STAT engines, however, to provide a degree of clearance control. It is estimated that blade tip clearances in this way may be held 15% smaller with the following impact on DOC:

Rated power--kW (hp)	% Δ Weight	% Δ Cost	% Δ sfc	% Δ DOC
1790 (2400)	-5.0	-8.0	-0.4	-1.141
3579 (4800)	-6.0	-6.0	-0.4	-1.094

Turbines

Advanced technology baseline engines were developed for examination of the trade-offs involved in choosing turbine and shafting arrangements compatible with high pressure ratio, axial, and axial-centrifugal compressors. The turbines were air-cooled to operate at 1506 K (2250°F), as was shown desirable in the previous section.

Configuration

Axial turbines were considered initially in the study for both size engines. Engine weights and prices were analytically determined by section, based on cycle descriptions, technology levels, and unique physical features. Cost data were estimated using the Material Index Factor (MIF) method employed by DDA.

1790 kW (2400 shp) STAT Engine

A baseline power section and eight variants were defined to compare different combinations of LP turbine shaft designs, Hp turbine materials, and methods of HP turbine blade attachments. The baseline configuration and variants are described in Table XXVII. The baseline configuration has a 20:1 CPR compressor at an airflow of 6.35 kg/s (14.0 lbm/sec) at approximately 38700 rpm. Tables XXVIII and XXIX show the comparative calculated weights by section, and also show calculated comparative recurring manufacturing prices.

TABLE XXVII. - 1790 kW (2400 shp) ENGINE FABRICATION TECHNOLOGY

Engine	Technology
Adv. Tech. Baseline	Composite, or hybrid, HP turbine wheels and blades (powdered metal discs and cast rim with integral blades), beryllium-reinforced LP turbine shaft
1	Forged IN-718 HP turbine wheels with dovetails and a beryllium-reinforced LP turbine shaft, 8% lower HP rpm
2	Near-net-shaped PA-101 HP turbine wheels with dovetails and a beryllium-reinforced LP turbine shaft, 8% lower HP rpm
3	Composite borsic titanium LP turbine shaft and composite HP turbine wheels without dovetails
4	Composite HP turbine wheels without dovetails and steel LP turbine shaft without beryllium insert (has mutes for damping)
5	Forged IN-718 HP turbine wheels with dovetails and a composite borsic titanium LP turbine shaft
6	Forged IN-718 HP turbine wheel with dovetails and a steel LP turbine shaft without beryllium insert (has mutes for damping)
7	Near-net-shaped HP turbine PA-101 wheels with dovetails and a composite LP turbine borsic titanium shaft
8	Near-net-shaped HP turbine PA-101 wheels with dovetails and a steel LP turbine shaft without beryllium insert (has mutes for damping)

TABLE XXVIII. - 1790 kW ENGINE POWER SECTION PRICES AND WEIGHT BREAKDOWN (SI UNITS)

	Adv. Tech. Baseline	1	2	3	4	5	6	7	8
Forward support--kg	11.45	12.93	12.93	11.45	11.45	11.45	11.45	11.45	11.45
Compressor rotor--kg	11.81	13.34	13.34	11.81	11.81	11.81	11.81	11.81	11.81
Compressor case--kg	19.56	22.10	22.10	19.56	19.56	19.56	19.56	19.56	19.56
Burner/diffuser--kg	27.59	31.18	31.18	27.59	27.59	27.59	27.59	27.59	27.59
HP turbine rotor	15.56	20.23	15.65	13.08	13.14	16.78	15.56	14.88	14.32
(wheels and spacer)--kg	(5.62)	(10.30)	(5.72)	(3.15)	(3.21)	(6.85)	(5.62)	(4.94)	(4.39)
HP turbine case--kg	9.18	10.37	10.37	9.18	9.18	9.18	9.18	9.18	9.18
LP turbine rotor (shaft)--kg	18.74	18.74	18.74	16.61	17.31	16.61	17.31	16.61	17.31
	(3.83)	(3.83)	(3.83)	(1.25)	(1.96)	(1.25)	(1.96)	(1.25)	(1.96)
LP turbine case--kg	9.17	9.17	9.17	9.17	9.17	9.17	9.17	9.17	9.17
Rear burner support--kg	11.34	11.34	11.34	11.34	11.34	11.34	11.34	11.34	11.34
Accy gearbox--kg	63.75	63.75	63.75	63.75	63.75	63.75	63.75	63.75	63.75
Total--kg	198.13	213.10	208.57	193.53	194.30	197.23	196.71	195.32	195.47
Recurring manufacturing price	$236,904	$260,660	$249,763	$235,547	$235,547	$249,389	$248,439	$240,025	$239,547

TABLE XXIX. - 2400 shp ENGINE POWER SECTION PRICES AND WEIGHT BREAKDOWN (CUSTOMARY UNITS)

	Adv. Tech. Baseline	1	2	3	4	5	6	7	8
Forward support--lbm	25.24	28.5	28.5	25.24	25.24	25.24	25.24	25.24	25.24
Compressor rotor--lbm	26.03	29.4	29.4	26.03	26.03	26.03	26.03	26.03	26.03
Compressor case--lbm	43.12	48.73	48.73	43.12	43.12	43.12	43.12	43.12	43.12
Burner-diffuser--lbm	60.83	68.74	68.74	60.83	60.83	60.83	60.83	60.83	60.83
HP turbine rotor	34.30	44.60	34.5	28.84	28.97	37.00	34.3	32.80	31.57
(wheels and spacer)--lbm	(12.4)	(22.7)	(12.6)	(6.94)	(7.07)	(15.1)	(12.4)	(10.9)	(9.67)
HP turbine case--lbm	20.24	22.87	22.87	20.24	20.24	20.24	20.24	20.24	20.24
LP turbine rotor (shaft)--lbm	41.31	41.31	41.31	36.61	38.17	36.61	38.17	36.61	38.17
	(7.46)	(7.46)	(7.46)	(2.76)	(4.32)	(2.76)	(4.32)	(2.76)	(4.32)
LP turbine case--lbm	20.21	20.21	20.21	20.21	20.21	20.21	20.21	20.21	20.21
Rear burner support--lbm	24.99	24.99	24.99	24.99	24.99	24.99	24.99	24.99	24.99
Accy gearbox--lbm	140.55	140.55	140.55	140.55	140.55	140.55	140.55	140.56	140.55
Total--lbm	436.8	469.80	459.81	426.66	428.35	434.82	433.68	430.62	430.95
Recurring manufacturing price	$236,904	$260,660	$249,763	$235,547	$235,547	$249,389	$248,439	$240,025	$239,347

The effect that the LP turbine shaft design changes had on the finished weight (FW) of the LP turbine and the FW of the HP turbine, due to varying shaft and wheel bore diameters, is shown for each variation in Table XXX. The effect that HP turbine wheel and blade attachment technology and reducing compressor rpm to improve LP turbine shaft critical speed had on FWs is also shown in Table XXX. It should be noted that an increase in the radius of the bore in the HP turbine resulting from the different LP turbine shaft designs greatly increases the FW of the wheels and spacer in the HP turbine because of the addition of bore reinforcing material. Note the increase in engine diameters as a result of slowing down the compressor rpm and the resultant FW penalty. However, the FW of the HP turbine wheels would have decreased with the reduced rpm in variations 1 and 2 were it not for the fact that nonintegral blades (attached by dovetails) increased the centrifugal load, stress, and FW of the HP turbine.

TABLE XXX. - 1790 kW (2400 shp) ENGINE EFFECT OF WHEEL SHAFT AND RPM VARIABLES ON SECTION WEIGHTS

Feature	Adv. Tech. Baseline	1	2	3	4	5	6	7	8
Wheels HP turbine									
Dovetails	Yes	Yes	Yes	No	No	Yes	Yes	Yes	Yes
Material	PA-101	IN-718	PA-101	PA-101	PA-101	IN-718	IN-718	PA-101	PA-101
% Δ weight	0	+83.2	+0.6	-44	-43	-22	0	-12.1	-22
R_b--mm (in.)	43.2	43.2	43.2	30.7	12.7	30.7	12.7	30.7	12.7
	(1.7)	(1.7)	(1.7)	(1.21)	(0.5)	(1.21)	(0.5)	(1.21)	(0.5)
Shafting									
Material	Stl/Be	Stl/Be	Stl/Be	Bo/Ti	Steel	Bo/Ti	Steel	Bo/Ti	Steel
% Δ weight	0	0	0	-63	-42	-63	-42	-63	-42
Supercritical	No	No	No	No	Yes	No	Yes	No	Yes
% Δ speed hp	0	-8.0	-8.0	0	0	0	0	0	0
% Δ HP turbine case FW	0	+13.3	+13.3	0	0	0	0	0	0
% Δ compressor case FW	0	+13.3	+13.3	0	0	0	0	0	0
% Δ burner/diffuser FW	0	+13.3	+13.3	0	0	0	0	0	0
% Δ compressor rotor FW	0	0*	0*	0	0	0	0	0	0

*Increased diameter and decreased speed cancelled out for no FW compared to baseline engine.

Table XXXI provides a breakdown of weights and price by engine section for variation 3 of the baseline 1790 kW (2400 shp) STAT engine. This is the winning configuration for both weight and price and provides the minimum DOC.

TABLE XXXI. - 1790 kW (2400 shp) (VARIATION 3) POWER SECTION AVERAGE RECURRING MANUFACTURING PRICE BY SECTION

(1000 engines at 12/month, in 1979 economics)

Engine section	FW--kg (lbm)	Baseline FW-- kg (lbm)	MIF_{ns}	I_c	Price--$	Baseline price--Δ$	Remarks
Forward support	11.45 (25.24)		6.26	0.158	7,291		
Axial compressor rotor	9.13 (20.13)		20.46	0.411	18,932		
Axial compressor case	11.13 (24.54)		17.76	0.435	20,072		
Centri compressor rotor	2.68 (5.90)		33.56	0.198	9,134		
Centri compressor case	8.43 (18.58)		15.09	0.280	12,935		
Burner/diffuser	27.59 (60.83)		8.88	0.540	24,931		
HP turbine rotor	13.08 (28.84)	-2.48 (-5.46)	21.90	0.631	27,951	-1764	Composite wheels
HP turbine case	9.18 (20.24)		14.25	0.288	11,788		without dovetails
LP turbine rotor	16.61 (36.61)	-2.13 (-4.7)	6.94	0.254	11,706	+ 407	Bo/Ti shaft
LP turbine case	9.17 (20.21)		7.09	0.143	6,609		
Rear burner support	11.34 (24.99)		9.06	0.226	10,450		
Accy gearbox	45.36 (100.00)						
Accessories	18.39 (40.55)		4.13	0.581	26,790		
Total	193.50 (426.6)	4.61 (-10.16)	(9.72)	4.146	*188,589	-1357	
Controls					40,715		
Assy. and Test (approx 90 hr)					6,243		
Average recurring manufacturing price					$235,547		

3579 kW (4800 shp) STAT Engine

The baseline engine established for the 3579 kW (4800 shp) size is an axial flow engine with a 20:1 CPR and an airflow of approximately 11.57 kg/s (25.5 lbm/sec) at approximately 29,800 rpm.

Five variations of the 3579 kW (4800 shp) baseline engine were selected: two variations of different combinations of HP turbine rpm, LP turbine shaft design, HP turbine materials, and two methods of blade attachments. Three variations have similar combinations as the first two but also have in addition an axial-centrifugal compressor and a shorter LP turbine shaft as a result of the combustor/diffuser design change. The technology describing the variations from baseline is shown in Table XXXII.

The comparative recurring manufacturing prices for the 3579 kW (4800 shp) engine power section, plus five variations in turbine design, were also prepared.

The calculated weights by section of the 3579 kW (4800 shp) baseline engine and five variations are shown in Tables XXXIII and XXXIV. Also shown are the calculated comparative recurring manufacturing prices of this engine and five variations.

TABLE XXXII. - 3579 kW (4800 shp) ENGINE FABRICATION TECHNOLOGY

Engine	Technology
Adv. Tech. Baseline	This engine is a straight through power section with IN-718 HP forged turbine wheels and individual dovetailed blades. In addition, the LP shaft is steel with no stiffening (i.e., beryllium). The shaft requires several damping mutes to lower the critical shaft speed below the operating range.
1	This variation is scaled from the baseline engine with a 2% decrease in HP rotor speed. The HP turbine wheels are made by powder metallurgy and include dovetails.
	The power turbine shaft is borsic titanium. The slight increase in HP turbine rotor weight, despite a reduced rpm, results from an increase in the bore diameter to pass the larger borsic titanium shaft required to get out of critical speeds.
2	This variation is the same as in 1 with the exception that the HP wheels and blades are composite without dovetails.
3	This variation is scaled from baseline with a 4% decrease in HP rotor speed. HP turbine wheels are forged IN-718 with dovetails with large ID. The LP turbine shaft is steel without mutes. The compressor is axial-centrifugal scaled from the 1790 kW (2400 shp) baseline engine.
4	This variation is the same as in 3 but it has an LP turbine shaft of borsic titanium. The HP rotor rpm is the same as that in the baseline.
5	This variation is the same as in 4 with the exception that it has HP turbine wheels and blades that are composite and steel LP shaft without mutes.

The LP turbine shaft design changes and their effects on FW of the LP turbine and FW of the HP turbine are shown in Table XXXV. The effect of HP turbine blade attachment and of reducing compressor rpm on FWs is also shown in Table XXXV. It should be noted that the baseline LP turbine shaft is "super critical." This condition is corrected by increasing the diameter of the different LP turbine shaft designs, which greatly increases the FW of the wheels and spacer in the HP turbine because of the increased radius of the bore reinforcing material. Note that the overall increase in engine diameter resulting from slowing down the compressor rpm, while maintaining the same R_c and W_a, has an FW penalty. The FW of the HP turbine wheels would have decreased with the reduced rpm in variations 2 and 4 were it not for the fact that the larger bore diameters increased the stress and FW of the HP turbine wheels.

TABLE XXXIII. - 3579 kW ENGINE POWER SECTION PRICES AND WEIGHT BREAKDOWN (SI UNITS)

	Adv. Tech. Baseline	1	2	3	4	5
Forward support--kg	20.70	21.37	20.76	22.04	20.70	20.70
Axial rotor--kg	24.89	24.89	24.89	20.57	20.57	20.57
Axial case--kg	30.35	31.34	30.35	27.81	26.10	26.10
Centrifugal rotor--kg				6.03	6.03	6.03
Centrifugal case--kg			9.66	9.07	9.07	
Burner/diffuser--kg	19.16	19.78		54.39	51.07	51.07
HP turbine rotor (wheels)--kg	21.77	22.47	20.57	26.26	22.58	17.28
	(9.98)	(10.68)	(8.75)	(14.47)	(10.80)	(5.49)
HP turbine case--kg	9.75	10.07	9.75	10.38	9.75	9.75
LP turbine rotor (shaft)--kg	21.04	19.29	19.29	20.17	17.00	20.14
	(6.02)	(4.27)	(4.27)	(5.15)*	(1.98)*	(5.15)*
LP turbine case--kg	24.11	24.11	24.11	24.11	24.11	24.11
Rear burner support--kg	18.21	18.21	18.21	18.21	18.21	18.21
Accessory--kg	62.11	62.11	62.11	62.11	62.11	62.11
Accessory gearbox--kg	14.70	14.70	14.70	14.70	14.70	14.70
Total--kg	266.8	268.3	263.8	316.4	302.0	299.8
Recurring manufacturer price	$401,591	$406,098	$397,793	$481,677	$467,943	$458,253

*Approximately 0.23 m shorter than baseline as a result of flow back burner configuration.

TABLE XXXIV. - 4800 shp ENGINE POWER SECTION AND WEIGHT BREAKDOWN (CUSTOMARY UNITS)

	Adv. Tech. Baseline	1	2	3	4	5
Forward support--lbm	45.63	47.113	45.63	48.60	45.63	45.63
Axial rotor--lbm	54.88	54.88	54.88	45.34	45.34	45.34
Axial case--lbm	66.91	69.09	66.91	61.30	57.53	57.53
Centrifugal rotor--lbm				13.29	13.29	13.29
Centrifugal case--lbm				21.3	20.0	20.0
Burner/diffuser--lbm	42.23	43.60	42.23	119.9	112.6	112.6
HP turbine rotor (wheels)--lbm	47.99	49.53	45.35	57.89	49.79	38.09
	(22.00)	(23.54)	(19.3)	(31.9)	(23.8)	(12.1)
HP turbine case--lbm	21.49	22.19	21.49	22.89	21.49	21.49
LP turbine rotor (shaft)--lbm	46.38	42.53	42.53	44.46	37.48	44.41
	(13.27)	(9.42)	(9.42)	(11.35)*	(4.37)*	(11.35)*
LP turbine case--lbm	53.16	53.16	53.16	53.16	53.16	53.16
Rear burner support--lbm	40.14	40.14	40.14	40.14	40.14	40.14
Accessory--lbm	136.92	136.92	136.92	136.92	136.92	136.92
Accessory gearbox--lbm	32.41	32.41	32.41	32.41	32.41	32.41
Total--lbm	588.1	591.6	581.6	697.6	665.8	661.0
Recurring manufacturer price	$401,591	$406,098	$397,793	$481,677	$467,943	$458,253

*Approximately 9 in. shorter than baseline as a result of flow back burner configuration.

TABLE XXXV. - 3579 kW (4800 shp) ENGINE STAT STUDY EFFECT OF WHEEL SHAFT AND RPM VARIABLES

Feature	Adv. Tech. Baseline	1	2	3	4	5
Wheels HP turbine						
Material	IN-718	PA-101	PA-101	IN-718	IN-718	PS-404
% Δ weight	0	+7.0	+12	+45	+8	-45
R_b--mm (in.)	16.5 (0.65)	59.7 (2.35)	59.7 (2.35)	50.8 (2.0)	36.8 (1.45)	50.8 (2.0)
Dovetails	Yes	Yes	No	Yes	Yes	Yes
Shafting LP turbine						
Material	Steel	Ti comp	Ti comp	Steel	Ti comp	Steel
% Δ weight	0	-29	-29	-14	-67	-14
Supercritical	Yes	No	No	No	No	No
R_o--mm (in.)	15.2 (0.6)	58.4 (2.30)	58.4 (2.30)	49.5 (1.95)	35.6 (1.40)	49.5 (1.95)
Length--m (in.)	1.04 (41)	1.04 (41)	1.04 (41)	0.81 (32)	0.81 (32)	0.81 (32)
% Δ HP speed	0	-2%	0	-4%	0	0
% Δ HP turbine case weight	0	+3.25%	0	+6.5%	0	
% Δ compressor case weight	0	+3.25%	0	+23.45%	+15.87%	+15.87%
% Δ burner/diffuser weight	0	+3.25%	0	+283.9%	+266.6%	+266.6%
% Δ compressor rotor	0	0	0	+6.83%	+6.83%	+6.83%

Table XXXVI provides a breakdown of weights and prices by engine section for variation 2 of the baseline 3579 kW (4800 shp) STAT engine. This combination of weight and price provided the minimum DOC.

TABLE XXXVI. - 3579 kW (4800 shp) (VARIATION 2) POWER SECTION AVERAGE RECURRING MANUFACTURING PRICE BY SECTION

(1000 engines at 12/month in 1979 economics)

Engine section	FW--kg (lbm)	Baseline FW-- kg (lbm)	MIF_{ns}	I_c	Price $	Baseline price--Δ$	Remarks
Forward support	20.70 (45.63)		4.47	0.204	9,406		Same as baseline
Axial compressor rotor	24.89 (54.88)		18.54	1.018	46,956		with the exception
Axial compressor case	30.35 (66.91)		16.27	1.090	50,242		of composite HP
Burner/diffuser	19.16 (42.23)		23.86	1.007	46,493		turbine wheels and
HP turbine rotor	20.57 (45.35)	-1.20 (-2.64)	32.35	0.969	43,440	-1805	blades and borsic
HP turbine case	9.75 (21.49)		21.88	0.469	19,573		titanium LP turbine
LP turbine rotor	19.29 (42.53)	-1.74 (-3.83)	15.78	0.671	30,970	+ 448	shaft
LP turbine case	24.11 (53.16)		13.29	0.706	32,612		
Rear burner support	18.21 (40.14)		7.89	0.317	14,522		
Accy gearbox	14.70 (32.41)		4.57	0.148	6,834		
Accessories	62.11 (136.92)		4.38	0.596	27,536		
Total	263.81 (581.6)	-2.95 (-6.5)	12.461	7.247	$328,584	-$1357	
Controls					54,287		
Assy. and Test (approx 180 hr)					12,485		
Average recurring manufacturer price					$395,357		

Turbine Flow Path

The turbine flow paths for the 1790 kW (2400 shp) and 3579 kW (4800 shp) STAT engines are shown in Figures 28 and 29, respectively. The appreciable step-up in diameter of the LP turbines' first stage, in both engines, necessitates a relatively long annulus between turbines and precludes the incorporation of counterrotating turbines.

The preliminary turbine designs for the STAT engine trade studies were made for the 1790 kW (2400 hp) engine at SLS intermediate power and were scaled in the radial direction by the square root of the ratio of compressor inlet airflows for the 3579 kW (4800 hp) engine. Meridional chords of the gas generator turbine were maintained when it was scaled up to the 3579 kW (4800 hp) size; however, the power turbine length was increased slightly to maintain an acceptable flare rate at the outer wall of the flow path.

The gas generator turbines were designed as two-stage axial flow units to keep stage equivalent work ($\Delta h/\theta_{cr}$) at an acceptable level of 54,638 J/kg (23.5 Btu/lbm). A single-stage transonic (high work) turbine would have resulted in a stage equivalent work of 101,603 J/kg (43.7 Btu/lbm) and a stage expansion ratio of 5.323 (supersonic) which is higher than DDA advanced design practice. The flow paths were designed for nearly constant hub diameter and have cylindrical, unshrouded rotor blade tips. Turbine average stage loading coefficient ($gJ\Delta h/Um^2$) was 1.6 and stage work was split 55 and 45% for the first and second stages, respectively. The maximum Mach numbers reached in the velocity diagrams were high subsonic.

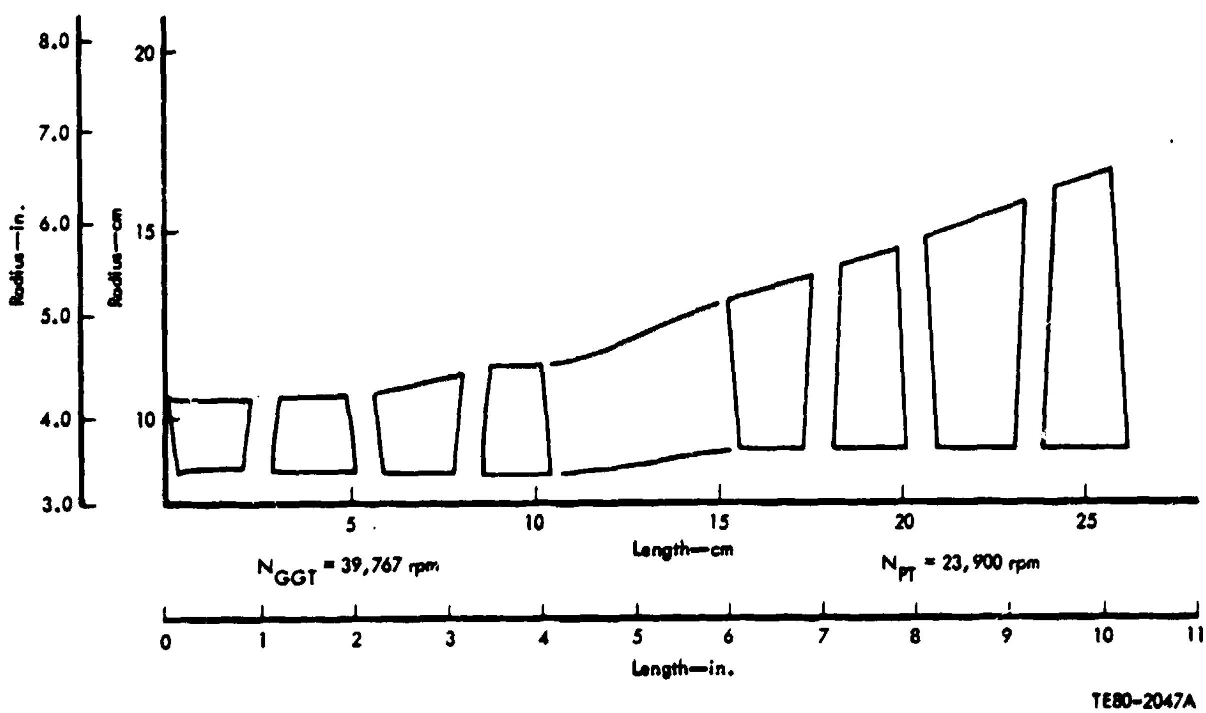


Figure 28. - STAT 1790 kW (2400 hp) engine turbine flow path.

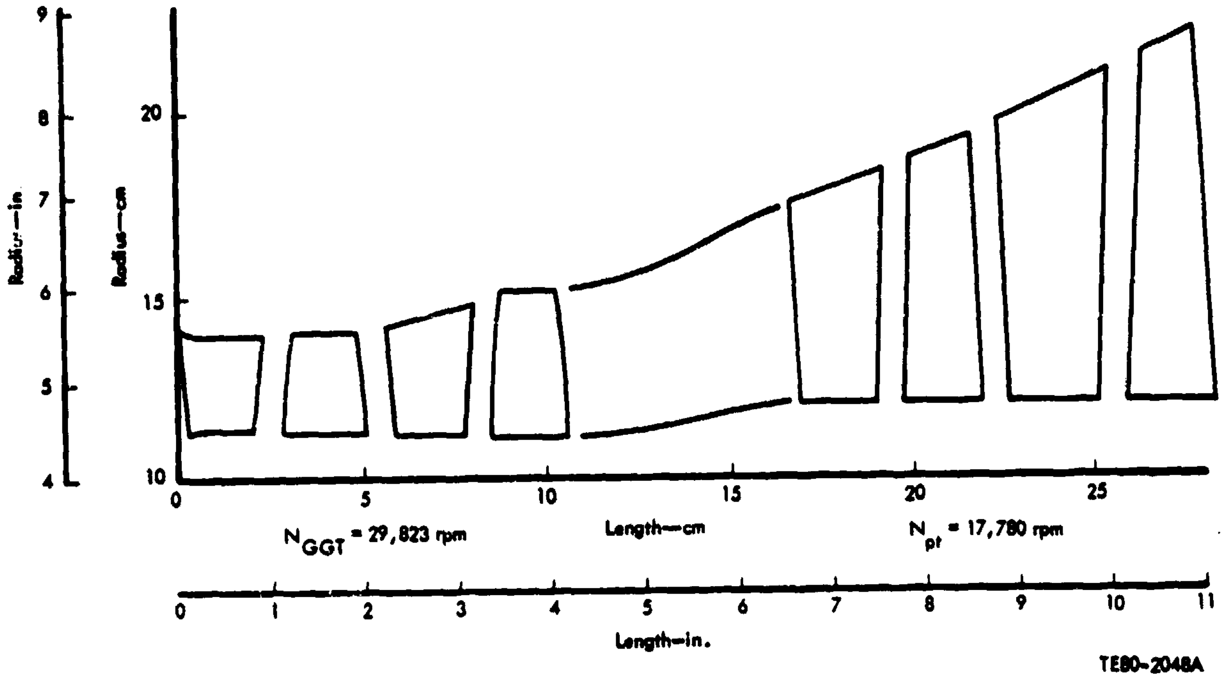


Figure 29. - STAT 3579 kW (4800 hp) engine turbine flow path.

The preliminary design study of the power turbines resulted in two-stage flow paths with constant hub diameter. Turbine exit area was sized to give an exit axial Mach number of 0.43 at the SLS point and the exit hub-to-tip diameter ratio was set at 0.54 to give adequate rotor blade hub reaction. Rotational speed was calculated to give a turbine exit AN^2 of about 5.4×10^{10}. The preceding design considerations directed the design to a two-stage flow path with an average stage loading coefficient of 1.6 to give the desired power turbine efficiency level. Power turbine work was split 55 and 45% for the first and second stages, respectively, to minimize turbine exit swirl at the altitude cruise point. A short transition duct was required to diffuse and transfer the gas flow outward from the gas generator turbine exit to the power turbine inlet.

A number of significant design problems are apparent for the advanced STAT engines. The problems believed to be most severe in our study of the power section were:

- o LP critical shaft speed and its impact on HP wheel design and lubrication system
- o Dovetail design limitations in small hub diameter engines
- o Design life (low cycle fatigue (LCF), stress rupture, oxidation/erosion) of short-range mission hardware
- o Minimum cooling air passage size set by the present casting state of the art
- o Airfoil size/height from a manufacturing standpoint and off-design sensitivity to tip clearance.

Radial Inflow Turbine

In order to investigate the potential performance improvement of a radial inflow turbine, such a flow path was configured (see Figure 30) for the 1790 kW (2400 hp) engine.

The results of this investigation show:

- o Axial engine length would not be significantly reduced (2 in. decreased) over that with an axial turbine.
- o Turbine efficiency is approximately the same (84-86%).
- o Small exit hub diameter precludes a concentric shaft engine design.

The desired sfc payoff was not realized, and this type of turbine was rejected for the STAT engines. The primary benefits of radial inflow turbines are best realized in small, low pressure ratio engines. A disadvantage for this type of turbine is that high rotor speeds are required for the radial inflow turbine which forces the compressor to run at a speed higher than optimum.

Hybrid Rotors, Composite Shafts, and Supercritical Shafts

In modern, two-spool concentric shaft gas turbines, the critical speed of the LP shaft influences engine size and configuration. This becomes a dominant limiting feature as engine pressure ratios increase, and airfoil hub diameters decrease, to maintain blade span.

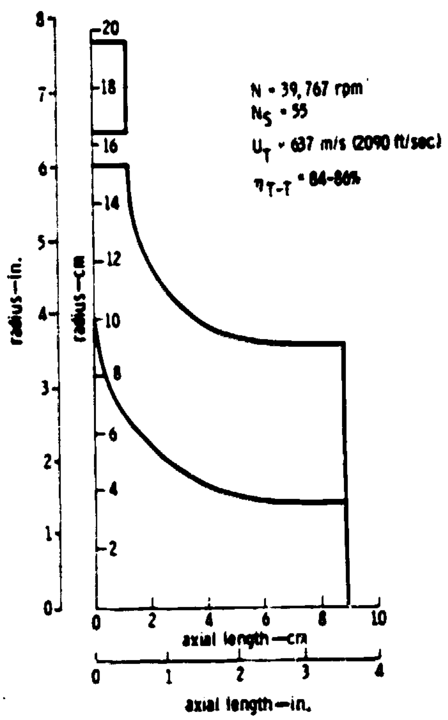


Figure 30. - STAT radial inflow turbine 1790 kW (2400 shp) engine.

This is a problem in the design of high pressure ratio engines, such as the 1790- and 3579 kW (2400- and 4800 shp) study engines. A trade study on shafting/wheel technology was undertaken. It is important to note wheel technology and shaft technology are integrated, because the HP spool wheels must have sufficient bore diameter to clear the LP shaft and any attachment features. Several terms used in this discussion are defined as follows:

Term	Features or material
Standard wheel	IN-718 wheels with dovetails
Advanced wheel	Power metallurgy wheel and dovetails
Composite (hybrid) wheel	Powder metallurgy wheel diffusion bonded to a cast blade ring
Ti-composite shaft	Titanium selectively strengthened with composite fibers
Supercritical	Used to describe shafting that, if simply supported, would have less than 25% speed between the first critical speed and the maximum operating speed. Some type of device, i.e., bearing, mute, etc., would be used to restrain motion.

As modern two-spool engine pressure ratios and RITs increase to improve sfc, the HP spool speed increases and its diameter decreases. The axial length tends to be fixed and independent of diameter.

A problem develops as this trend continues. Soon the ability of the designer to use dovetails for blade retention on the wheels, and to use conventional, simply supported steel shafting for the LP shaft, is restricted. In the preliminary flow paths referred to as the advanced baseline engines, it is not possible to configure a conventional technology wheel (conventional material and dovetails) around the minimum OD subcritical shaft made of steel.

A number of technology and configuration trades were conducted in the course of this study and resulting weight penalties were assessed. The following items were considered:

- Wheels
  - Enhanced properties with dovetails
  - Composite or HIP-bonded wheel/blade assemblies
- Shafting
  - Composite or fiber-reinforced titanium
  - Supercritical shafting
- General arrangement and speed
  - Close coupled LP turbine
  - Decreased HP spool speed

The final evaluations for critical shaft speed, with resulting impacts on DOC, are given in Tables XXXVII through XL for the 3579 kW and 1790- (4800- and 2400 hp) STAT engines, respectively. It will be seen that the combination of technologies for 1790 kW (2400 shp) engine No. 7 provides the biggest payoff in DOC improvement. This engine incorporates composite wheels and shafts and cast-in impingement cooling.

TABLE XXXVII. - 3579 kW SHAFT STUDY RESULTS (SI UNITS)

1979 economics

	PD370-37 scaled	STAT baseline							
	Base	1	2	3	4	5	6	7	8
Cycle									
R_c	12.7	20:1	20:1	20:1	20:1	20:1	20:1	20:1	20:1
RIT--K	1505	1505	1505	1505	1505	1505	1505	1505	1505
SFC--Mg/W·s	79.75	66.23	66.23	66.23	66.40	66.23	67.42	67.42	67.42
Power section									
Mass--kg	381.5	266.8	268.3	268.3	268.3	263.8	293.7	302.0	399.8
Cost--$	276,752	262,590	265,514	264,783	264,607	260,128	313,170	296,727	291,080
Length--m	1.466	1.466	1.466	1.466	1.466	1.466	1.237	1.237	1.237
Diameter--m	0.503	0.495	0.495	0.495	0.495	0.495	0.759	0.759	0.759
Gearbox									
Mass--kg	156.0	156.0	156.0	156.0	156.0	156.0	156.0	156.0	156.0
Cost--$	59,000	59,000	59,000	59,000	59,000	59,000	59,000	59,000	59,000
Length--m	0.255	0.255	0.255	0.255	0.255	0.255	0.255	0.255	0.255
Ratio	10.4	10.4	10.4	10.4	10.4	10.4	10.4	10.4	10.4
Totals									
Mass--kg	537.5	422.8	424.4	424.4	424.4	419.8	449.8	458.0	455.9
Cost--$	335,752	321,590	324,514	323,783	323,607	319,128	372,170	355,727	350,080
Length--m	2.644	2.644	2.644	2.644	2.644	2.644	2.416	2.416	2.416
Diameter--m	0.503	0.495	0.495	0.495	0.495	0.495	0.759	0.759	0.759
% Δ weight		-21.3	-21.1	-21.6	-21.6	-21.9	-16.3	-14.8	-15.2
% Δ cost		-4.2	-3.3	-3.6	-3.6	-4.95	+10.85	+5.95	+4.27
% Δ length		0	0	0	0	0	-8.6	-8.6	-8.6
% Δ diameter		-1.5	-1.5	-1.5	-1.5	-1.5	+50.9	+50.9	+50.9
% Δ SFC		-16.9	-16.9	-16.9	-16.7	-16.9	-15.5	-15.5	-15.5
% Δ DOC		-7.16	-7.04	-7.09	-7.02	-7.27	-2.67	-3.25	-3.48
Technology									
Configuration		Axial	Axial	Axial	Axial	Axial	Axial-cent	Axial-cent	Axial-cent
HP turbine wheels		Forged Super	Net shape	Net shape	Net shape	Composite	Forged	Forged	Composite
Shafting		Super	Composite	Composite	Composite	Composite	Steel	Composite	Steel
First blade cooling		Imp/film cast-in	Imp/film tube	Imp/film cast-in	Convection	Imp/film tube	Imp/film cast-in	Imp/film cast-in	Imp/film cast-in

The study of 3579 kW (4800 hp) engines did not encompass as many combinations of technologies as did that for the 1790 kW (2400 hp) engines. Engine No. 5 exhibited the best DOC improvement with composite shafts and wheels and tube-type impingement cooling. If this engine had cast-in impingement cooling, it would be expected that the percent change in DOC would be improved an additional 0.05%. This is substantiated by comparing the percent change in DOC for Engine No. 2 and No. 3, which differ in configuration only in the type of first blade cooling.

TABLE XXXVIII. - 4800 shp SHAFT STUDY RESULTS (CUSTOMARY UNITS)

1979 economics

	PD370-37 scaled	STAT baseline							
	Base	1	2	3	4	5	6	7	8
Cycle									
R_c	12.7	20:1	20:1	20:1	20:1	20:1	20:1	20:1	20:1
RIT--°F	2250	2250	2250	2250	2250	2250	2250	2250	2250
SFC	0.472	0.392	0.392	0.392	0.393	0.392	0.399	0.399	0.399
Power section									
Weight--lbm	841.0	588.1	591.6	591.6	591.6	581.6	647.6	665.8	661.0
Cost--$	276,752	262,590	265,514	264,783	264,607	260,128	313,170	296,727	291,080
Length--in.	57.7	57.7	57.7	57.7	57.7	57.7	48.7	48.7	48.7
Diameter--in.	19.81	19.5	19.5	19.5	19.5	19.5	29.9	29.9	29.9
Gearbox									
Weight--lbm	344	344	344	344	344	344	344	344	344
Cost--$	59,000	59,000	59,000	59,000	59,000	59,000	59,000	59,000	59,000
Length--in.	10.02	10.02	10.02	10.02	10.02	10.02	10.02	10.02	10.02
Ratio	10.4	10.4	10.4	10.4	10.4	10.4	10.4	10.4	10.4
Totals									
Weight--lbm	1185	932.1	935.6	935.6	935.6	925.6	991.6	1009.8	1005.0
Cost--$	335,752	321,590	324,514	323,783	323,607	319,128	372,170	355,727	350,080
Length--in.	104.1	104.1	104.1	104.1	104.1	104.1	95.1	95.1	95.1
Diameter--in.	19.81	19.5	19.5	19.5	19.5	19.5	29.9	29.9	29.9
% Δ weight		-21.3	-21.1	-21.6	-21.6	-21.9	-16.3	-14.8	-15.2
% Δ cost		-4.2	-3.3	-3.6	-3.6	-4.95	+10.85	+5.95	+4.27
% Δ length		0	0	0	0	0	-8.6	-8.6	-8.6
% Δ diameter		-1.5	-1.5	-1.5	-1.5	-1.5	+50.9	+50.9	+50.9
% Δ SFC		-16.9	-16.9	-16.9	-16.7	-16.9	-15.5	-15.5	-15.5
% Δ DOC		-7.16	-7.04	-7.09	-7.02	-7.27	-2.67	-3.25	-3.48
Technology									
Configuration		Axial	Axial	Axial	Axial	Axial	Axial-cent	Axial-cent	Axial-cent
HP turbine wheels		Forged	Net shape	Net shape	Net shape	Composite	Forged	Forged	Composite
Shafting		Super	Composite	Composite	Composite	Composite	Steel	Composite	Steel
First blade cooling		Imp/film cast-in	Imp/film tube	Imp/film cast-in	Convection	Imp/film tube	Imp/film cast-in	Imp/film cast-in	Imp/film cast-in

An additional benefit is available for turbine blades with cast-in impingement cooling passages, as a greater percentage of the wheel rim is available for blade retention. This is due to the smaller blade dovetail and resultant increase in spacing of dovetail slots in the wheel rim.

As engine size decreases, the practicality of using dovetail blade attachments, in both compressor and turbine, decreases. In the case of our advanced baseline engines, the presence of cooling features (i.e., impingement-cooled first blade) complicates the picture. A preliminary dovetail was sized for the first turbine blade; however, space for impingement tube passage is minimal.

Thermal Barrier Coatings

Thermal barrier coatings show greatest promise in very high temperature engines. Their effect is greatest where high gas side-to-metal temperature gradients are required.

TABLE XXXIX. - 1790 kW SHAFT STUDY RESULTS (SI UNITS)

1979 economics

	PD370-37 scaled	STAT baseline												
	Base	1	2	3	4	5	6	7	8	9	10	11	12	13
Cycle														
R_c	12.7	20	20	20	20	20	20	20	20	20	20	20	20	20
RIT--°F	2250	2250	2250	2250	2250	2250	2250	2250	2250	2250	2250	2250	2250	2250
sfc	0.5098	0.435	0.435	0.436	0.435	0.436	0.435	0.435	0.436	0.435	0.435	0.435	0.435	0.435
Power section														
Weight--lbm	578.5	437.0	469.5	469.5	459.4	459.4	426.6	426.6	426.6	420.4	434.8	433.6	430.6	430.9
Cost--$	155,681	155,372	168,489	168,313	161,414	161,238	154,493	153,965	153,788	154,493	162,941	162,324	156,867	156,429
Length--in.	43.80	36.0	36.0	36.0	36.0	36.0	36.0	36.0	36.0	36.0	36.0	36.0	36.0	36.0
Diameter--in.	14.01	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4
Gearbox														
Weight--lbm	301.4	301.4	301.4	301.4	301.4	301.4	301.4	301.4	301.4	301.4	301.4	301.4	301.4	301.4
Cost--$	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735
Length--in.	8.602	8.602	8.602	8.602	8.602	8.602	8.602	8.602	8.602	8.602	8.602	8.602	8.602	8.602
Totals														
Weight--lbm	879.9	738.4	770.9	770.9	760.8	760.8	728.0	728.0	728.0	729.8	736.2	735.0	732.0	732.4
Cost--$	209,616	209,107	222,224	222,048	215,149	214,973	208,228	207,700	207,523	208,228	216,676	216,059	210,602	210,164
Length--in.	84.73	76.93	76.93	76.93	76.93	76.93	76.93	76.93	76.93	76.93	76.93	76.93	76.93	76.93
Diameter--in.	14.01	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4	23.4
% Δ weight		-16.0	-12.4	-12.4	-13.5	-13.6	-17.3	-17.3	-17.3	-17.1	-16.3	-16.5	-16.8	-16.8
% Δ cost		-0.24	+6.0	+5.9	+2.6	+2.6	-0.66	-0.91	-1.0	-0.66	+3.4	+3.1	+0.47	+2.61
% Δ length		-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2
% Δ diameter		+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0
% Δ sfc		-14.67	-14.67	-14.48	-14.67	-14.48	-14.67	-14.67	-14.48	-14.67	-14.67	-14.67	-14.67	-14.67
% Δ DOC		-4.88	-4.11	-4.04	-4.51	-4.44	-4.97	-4.993	-4.924	-4.96	-4.50	-4.54	-4.83	-4.60
Technology														
HP wheels		Composite	Forged	Forged	Net shape with dovetail	Net shape with dovetail	Composite	Composite	Composite	Composite	Forged	Forged	Near net shape	Near net shape
Shaft		Steel	Steel	Steel	Steel	Steel	Composite	Composite	Composite	Super	Composite	Super	Composite	Super
First blade cooling		Imp/film tube	Imp/film cast-in	Convection	Imp/film cast-in	Convection	Imp/film tube	Imp/film cast-in	Convection	Imp/film tube	Imp/film cast-in	Imp/film cast-in	Imp/film cast-in	Imp/film cast-in

TABLE XL. - 2400 shp SHAFT STUDY RESULTS (CUSTOMARY UNITS)

1979 economics

	PD370-37 scaled	STAT baseline												
	Base	1	2	3	4	5	6	7	8	9	10	11	12	13
Cycle														
R_c	12.7	20	20	20	20	20	20	20	20	20	20	20	20	20
RIT--K	1505	1505	1505	1505	1505	1505	1505	1505	1505	1505	1505	1505	1505	1505
sfc--Ng/W·s	86.139	73.50	73.50	73.67	73.50	73.67	73.50	73.50	73.67	73.50	73.50	73.50	73.50	73.50
Power section														
Mass--kg	262.4	198.2	213.0	213.0	208.4	208.4	193.5	193.5	193.5	194.3	197.2	196.7	195.3	195.5
Cost--$	155,681	155,372	168,489	168,313	161,414	161,238	154,493	153,965	153,788	154,493	162,941	162,324	156,867	156,429
Length--m	1.112	0.914	0.914	0.914	0.914	0.914	0.914	0.914	0.914	0.914	0.914	0.914	0.914	0.914
Diameter--m	0.356	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594
Gearbox														
Mass--kg	136.7	136.7	136.7	136.7	136.7	136.7	136.7	136.7	136.7	136.7	136.7	136.7	136.7	136.7
Cost--$	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735	53,735
Length--m	0.218	0.218	0.218	0.218	0.218	0.218	0.218	0.218	0.218	0.218	0.218	0.218	0.218	0.218
Totals														
Mass--kg	399.1	334.9	349.7	349.7	345.1	345.1	330.2	330.2	330.2	331.6	333.9	333.4	332.0	332.2
Cost--$	209,616	209,107	222,224	222,048	215,149	214,973	208,228	207,700	207,523	208,228	216,676	216,059	210,602	210,164
Length--m	2.152	1.954	1.954	1.954	1.954	1.954	1.954	1.954	1.954	1.954	1.954	1.954	1.954	1.954
Diameter--m	0.356	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594	0.594
% Δ weight		-16.0	-12.4	-12.4	-13.5	-13.6	-17.3	-17.3	-17.3	-17.1	-16.3	-16.5	-16.8	-16.8
% Δ cost		-0.24	+6.0	+5.9	+2.6	+2.6	-0.66	-0.91	-1.0	-0.66	+3.4	+3.1	+0.47	+2.61
% Δ length		-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2	-9.2
% Δ diameter		+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0	+67.0
% Δ sfc		-14.67	-14.67	-14.48	-14.67	-14.48	-14.67	-14.67	-14.48	-14.67	-14.67	-14.67	-14.67	-14.67
% Δ DOC		-4.88	-4.11	-4.04	-4.51	-4.44	-4.97	-4.993	-4.924	-4.96	-4.50	-4.54	-4.83	-4.60
Technology														
HP wheels		Composite	Forged	Forged	Net shape with dovetail	Net shape with dovetail	Composite	Composite	Composite	Composite	Forged	Forged	Near net shape	Near net shape
Shaft		Steel	Steel	Steel	Steel	Steel	Composite	Composite	Composite	Super	Composite	Super	Composite	Super
First blade cooling		Imp/film tube	Imp/film cast-in	Convection	Imp/film cast-in	Convection	Imp/film tube	Imp/film cast-in	Convection	Imp/film tube	Imp/film cast-in	Imp/film cast-in	Imp/film cast-in	Imp/film cast-in

An analysis of cooling flows in the 1790 kW (2400 shp) STAT engine indicates the coating would reduce chargeable cooling by 20%; however, the increase in airfoil thickness reduces turbine efficiency due to the blockage effect. Using sensitivity factors for DOC versus cooling airflow and turbine efficiency, the following negative impact is shown:

	% Δ DOC
-1% η HP turbine	+1.25
-20% Δ cooling	-1.10
Net impact	+0.15% DOC

A similar loss characteristic would be experienced by the 3579 kW (4800 shp) STAT engine.

The assumptions used in this analysis are:

- o The blade life is constant if metal temperature is held constant. Here we chose not to evaluate the effect of increased blade load through the ceramic mass.
- o A constant thickness coating was assumed over the entire airfoil.
- o The effect of changing coating thickness through erosion was not accounted for in life calculations.
- o The surface finish of the coating is the same as that for the uncoated blade.

It should be noted that independent industry test programs have shown that turbine airfoil coatings in the unpolished state tend to have a rougher finish than the parent metal. In some cases, this rougher finish can cause the reverse effect of making the airfoils hotter with the coating than without.

Ceramics

DDA is involved in programs to develop ceramics for gas turbine engines. These efforts are directed to two areas: solid, monolithic structures, and coatings. The solid, monolithic structures, such as turbine vanes and blades, show the greatest promise in reducing airfoil cooling airflows. For our STAT advanced technology engines, total chargeable cooling airflows could drop by 4.5% of engine airflow. This is a 45% reduction from the advanced baseline cycles.

Within the time frame specified for STAT engines, however, monolithic ceramic components were not considered due to current state of the art plus normal time for design, procurement and test ahead of full development release in 1988.

Ceramic coatings have reached a higher level of development than monolithic structures. Coatings for airfoils and end-wall insulation have been used in some advanced experimental engines at DDA. In addition, an abradable ceramic coating has been tried for first- and second-stage tip shrouds. The coating used was eroded severely in less than 35 hours of engine operation. The vane airfoil coatings have had some success in experimental engine operation; however, the resistance to environmental particulates needs improvement. Ceramic coatings were, therefore, not considered.

Cast-in-Impingement Cooling

Cooling air distribution through turbine blades by means of transpiration via Lamilloy is very effective where the airfoils are large enough for trailing edge blockage effects to be minimal. The airfoils in the STAT engines are too small for this consideration, and employ impingement cooling instead.

Presently, all DDA production air-cooled airfoils (blades or vanes) are made by an investment casting process using cores to form the internal passages. At present the minimum core size is limited by the strength of the core material. In addition, some air-cooled airfoils use impingement cooling to increase cooling effectiveness. The small size of some airfoils makes it difficult to manufacture and assemble sheet metal tubes small enough to fit within them.

One method of circumventing the tube problem is to cast-in the impingement airflow passage, as shown in Figure 31. This is done by inserting quartz rods between core sections. The rods are removed from the casting chemically, leaving the required impingement holes. This method has the advantage of providing more load-bearing metal area within the same airfoil contour. A simple costing study indicates a $595 savings per stage for the small engine, and $741 for the 3579 kW (4800 shp) engine. Since the first vanes and second vanes will most likely require impingement cooling, the cost reductions become:

Rated power--kW (shp)	1790 (2400)	3579 (4800)
$ Δ engine cost	1190	1482
% Δ engine cost	-0.580	-0.494
% Δ DOC	-0.063	-0.064

There should be additional savings to the engine via lowered cooling flows and improved life inherent with this configuration. This results from having a structural load-carrying member isolated from the gas side temperatures.

Long Life Bearings

The main bearings incorporated throughout the STAT engines will exhibit a much greater load carrying capability and fatigue life than those in current technology engines. This improvement may be brought about in part by research in the characterization and control of forging flow lines and end grain areas in bearing balls.

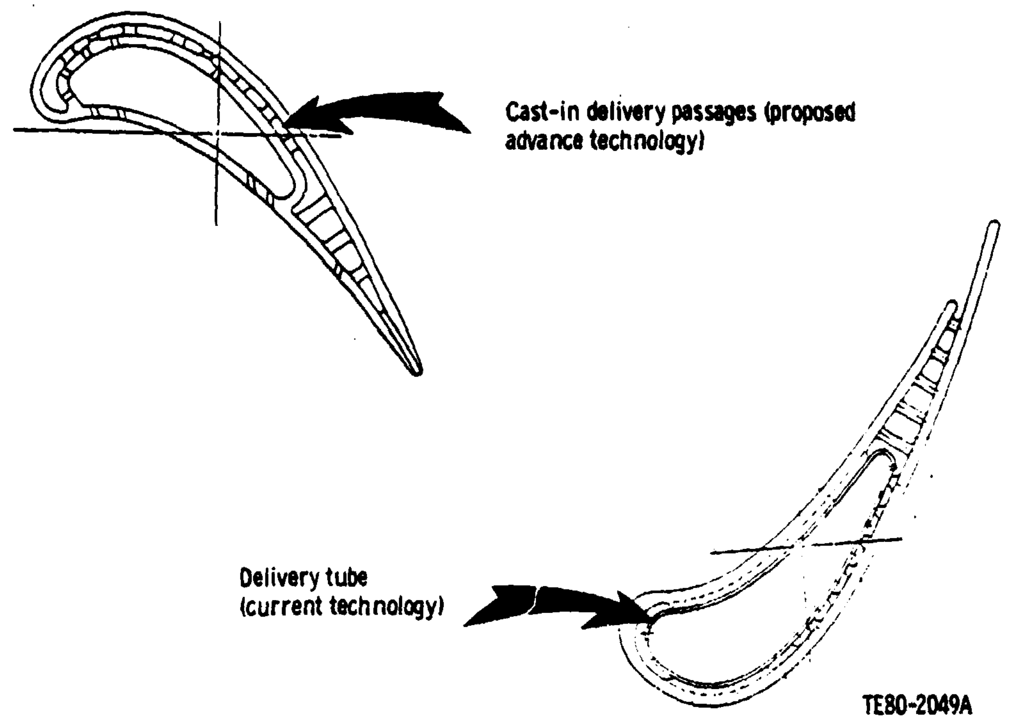


Figure 31. - Airfoil impingement cooling.

In the case of the STAT reduction gear main bearings, DDA studies show that an improvement in the inherent premature removal rate of approximately 57% is possible by this method. This equates to a reduction in engine maintenance cost of 1.1%, and resulting decrease in DOC of 0.11% for the 1790 kW (2400 hp) STAT engine, and 0.14% for the 3579 kW (4800 hp) engine.

Clearance Control

If the STAT HP turbines were straddle-mounted, the second stage clearance would reduce by 0.0013 in. and the first by 0.0008 in. The increased weight and cost of the required bearing support would penalize the engine. The following table assesses the penalties and payoffs:

Rated power--kW (hp)	% Δ Weight	% Δ Cost	% Δ sfc	% Δ DOC
1790 (2400)	+3%	+3.6%	-.31%	+.345
3579 (4800)	+3.6%	+3.1%	-.31%	+.399

There is, therefore, a DOC penalty rather than an improvement, if the HP turbines are straddle-mounted.

The most critical feature required of an active clearance control system for a short-haul aircraft is a quick response to transients. Systems that do not respond rapidly to transients, such as the general class of thermally actuated systems, will not be cost effective in small transport aircraft.

Most active clearance control systems considered previously have mechanically driven tip seals. For this study, consideration has been given to a simple system to axially translate sloped tip seals as shown in Figure 32. This device would be activated by the electronic control and would be continuously adjusted to maintain the safe, minimum clearance during the entire mission.

The turbine flow path for a current gas turbine under development at DDA is similar to both the 3579- and 1790 kW (4800- and 2400 shp) STAT engines with respect to shape and blade height. The blade heights of the first and second turbine stages of this engine are approximately 1.32 and 2.13 cm (0.52 and 0.84 in.), respectively. Using the general relationship that for each change in clearance equal to 1% of blade height, a 2% change in turbine stage efficiency results, then gasifier turbine efficiency changes 1% for each 0.13 mm (0.0052 in.) change in first-stage clearance, or 0.21 mm (0.0084 in.) change in second-stage clearance.

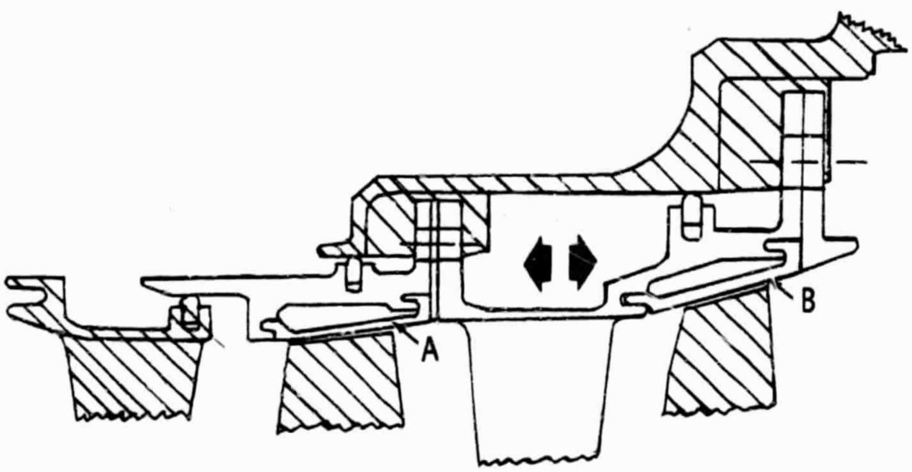


Figure 32. - Active clearance control--turbines.

The clearance of the first blade is smallest during start and warmup. This results in approximately 0.16 mm (0.0062 in.) additional clearance during normal running operation, and a 1.2% loss in gasifier turbine efficiency. The second-stage blade clearance is also least during start, resulting in an additional 0.19 mm (0.0074 in.) change in clearance. This penalized the gasifier turbine efficiency an additional 0.88%. Total gasifier payoff is thus estimated to be 2.08%. If we assume that an active clearance control system would eliminate this penalty, the sfc would improve by 2.3%, and DOC would decrease by 0.89% if the weight and cost penalties are not included.

Based on approximations of system weight and cost, the following results would be obtained for the STAT engines:

Rated power--kW (hp)	% Δ Weight	% Δ Cost	% Δ sfc	% Δ DOC
1790 (2400)	+1.7	+5.7	-2.3	-0.284
3579 (4800)	+1.6	+4.8	-2.3	-0.137

As an alternate to an active clearance control, thermal matching offers a lower risk, albeit lower payoff means of clearance control. DDA development experience has shown that increased attention must be paid to transient response of rotor and stationary elements. Specifically, the HP rotor case material was changed for a current DDA development engine, and the transient-induced running clearances were reduced from 0.16 to 0.06 mm (0.0062 to 0.0025 in.) for the first stage, and from 0.19 to 0.13 mm (0.0074 to 0.005 in.) for the second stage.

Given sfc sensitivity to turbine clearance, such reductions in the STAT engines should improve sfc by 1%, reduce DOC by 0.35, and 0.31% for 3579- and 1790 kW (4800- and 2400 shp) engines, respectively.

Leakage Control

The STAT engine turbines will use paired step-seals to minimize air leakage and stage bypass flow. This technology promises a 10 to 20% reduction in flow for the same number of seal elements. The primary technology advancement required to achieve this benefit lies in the modeling of the labyrinth seal flows analytically instead of through experimental test.

Abradable Turbine Coatings

Two reasons for using abradable coatings in the turbine blade tip path have been identified. The first is in making the engine more tolerant of turbine rotor offset caused by normal production tolerances. The second is to minimize requirements allowing for extreme transient conditions, thus increasing safety because of the ability to have extreme transients and not experience major structural damage. These reasons apply whether the turbine incorporates active clearance control features or not.

A current DDA development engine incorporates abradable turbine seals. This engine would be expected to experience 0.019 negative growth during an emergency shutdown. If the blades were running with a nonabradable seal surface, they would most likely be severely damaged. The additional clearance running, to avoid shutdown damage, causes an unacceptable sfc penalty to the engine. Incorporation of abradable seals, however, could provide the following typical improvement in DOC:

Rated power--kW (hp)	% Δ Weight	% Δ Cost	% Δ sfc	% Δ DOC
1790 (2400)	-5.0	-8.0	-0.8	-1.046
3579 (4800)	-6.0	-6.0	-0.8	-1.212

Two methods are under consideration for the use of abradable coatings—to develop a truly abradable coating for conventional blades, or to combine a partly abradable coating with abrasive tip blades. The state of development in this area would indicate this is a high risk technology for commercial engine development.

One method currently used to maintain close tolerance turbine blade tip clearance is to line-bore the rotor bearing support cavities with the structural members assembled sans the rotor. This is a costly process which could be eliminated if abradable seals were available. An additional payoff for abradable coatings is the fact that safety improves without sacrificing performance. In both STAT engines, the penalty in performance to design for centerline offset and emergency shutdown could not be accepted.

Design Life

The standard practice for turbine blades is to coat them with materials to provide protection from sulfidation and other forms of corrosion, and thereby assure long life. The limiting feature of the hot section, as far as overhaul time, is the life of the blade coating. Based on available materials data, an airfoil will require two to three recoats during its life.

A study of first-stage turbine blade life was conducted. An analysis of cooling air requirement sensitivity to changes in pressure ratio, RIT, and cooling air temperature is shown in Figure 33. The impact of cooling airflow on life of the first-stage blade is shown in Figure 34.

A study was also made of the impact of increasing turbine wheel strength (weight) on LCF life. This sensitivity is depicted on Figure 35. It should be noted that no design restraints have been considered in limiting the size of the wheel while increasing its weight.

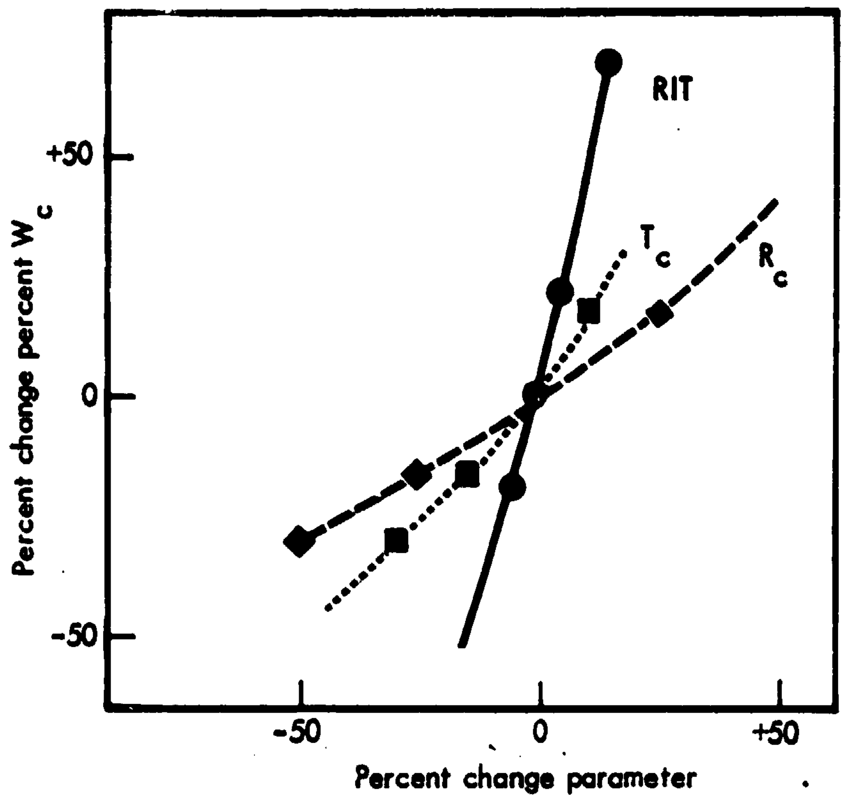


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Figure 33. - STAT first-stage blade sensitivity of cooling flow to turbine temperature, pressure ratio, and cooling air temperature.

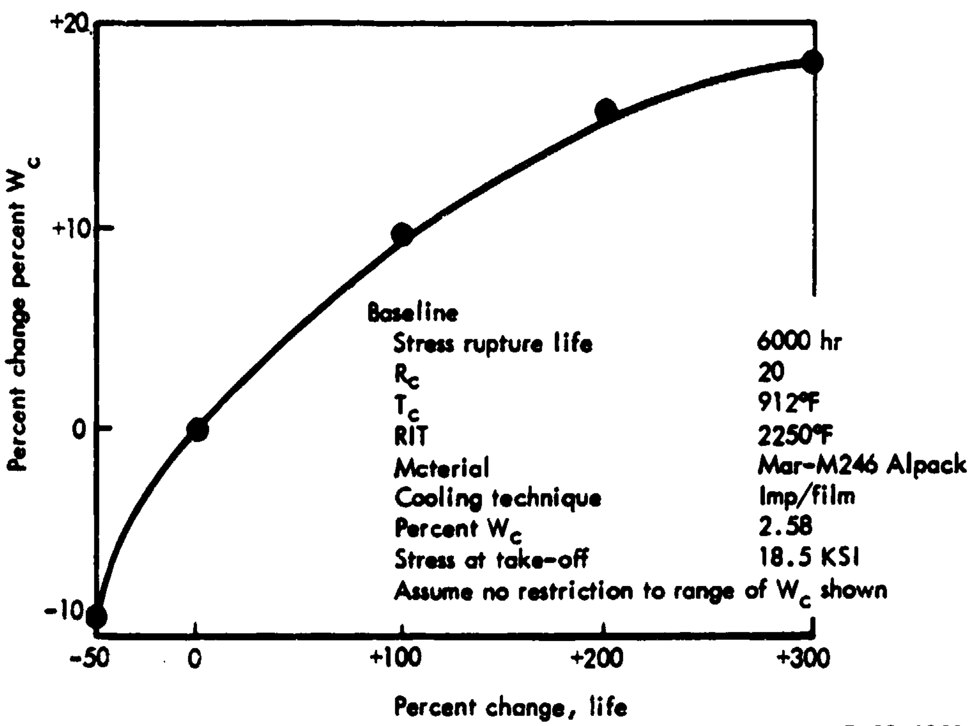

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Figure 34. - STAT first-stage blade sensitivity of cooling flow to life.

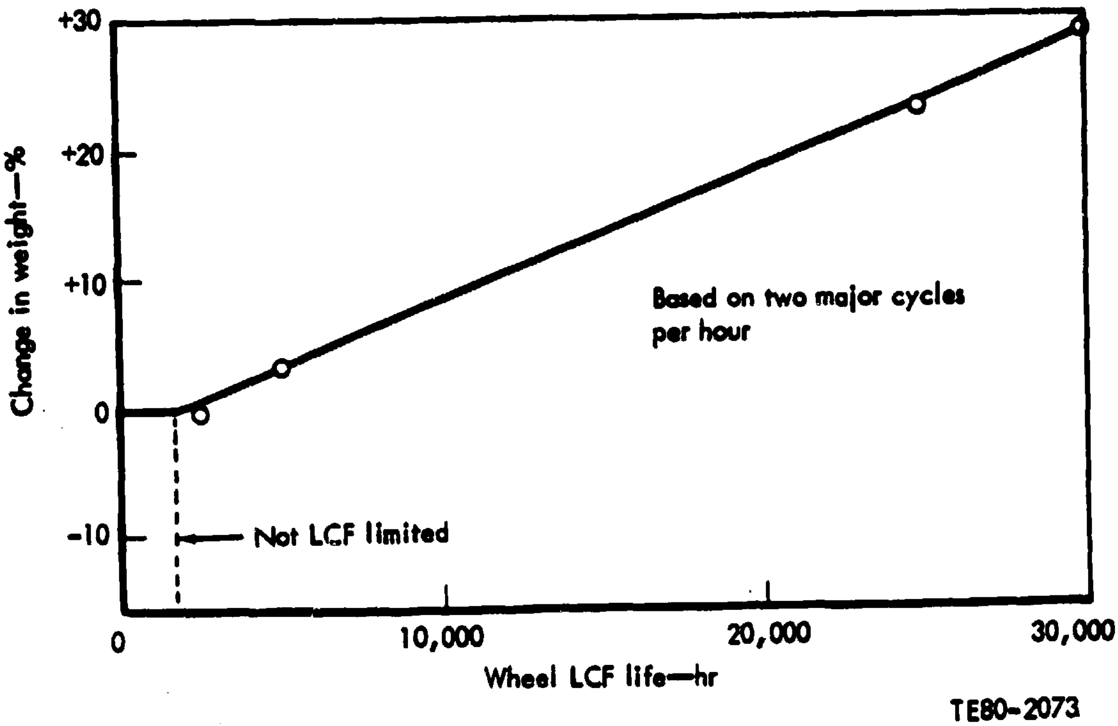


Figure 35. - STAT wheel life versus percent change in wheel weight.

Bearings

Tapered Roller Bearings

Tapered roller bearings have been suggested as substitutes for conventional ball thrust bearings used in most modern gas turbine engines. These bearings have calculated life improvements of at least 50% above the ball bearings they replace. However, it is necessary to install matched pairs of tapered roller bearings in the place of a single ball bearing with the attendant increase in cost, parts inventory, maintenance effort, etc. Therefore, it is judged that DOC would not benefit appreciably from this technology.

Beryllium-Backed Bearing Races

In our STAT engine sizes, beryllium-backed bearing races would have a small payoff since the primary benefit is weight reduction. In large engine gearboxes, the bearings have more dominant design limitations, and a stronger payoff would be expected.

Diffusers/Combustors

Combustor flow paths were established in both STAT engine sizes by scaling existing combustors. Both engines have annular combustors produced with *Lamilloy® technology. The large engine uses a straight-through annular combustor. The small engine has a fold back design that takes advantage of a decrease in engine length available with no increase in power section frontal area. This is possible due to its compressor discharge being radially outward and not axial.

Several diffuser and combustor advance technologies were suggested as trade study elements. Of these, the vortex-controlled diffuser (VCD) and transpiration-cooled Lamilloy combustor were utilized since they were the subject of recent DDA life cycle cost studies under Air Force contract No. F33657-77-C-0425. A description of these technologies and their impact on engine weight, cost, performance, etc. is contained in the final report for this USAF contract.(1)**

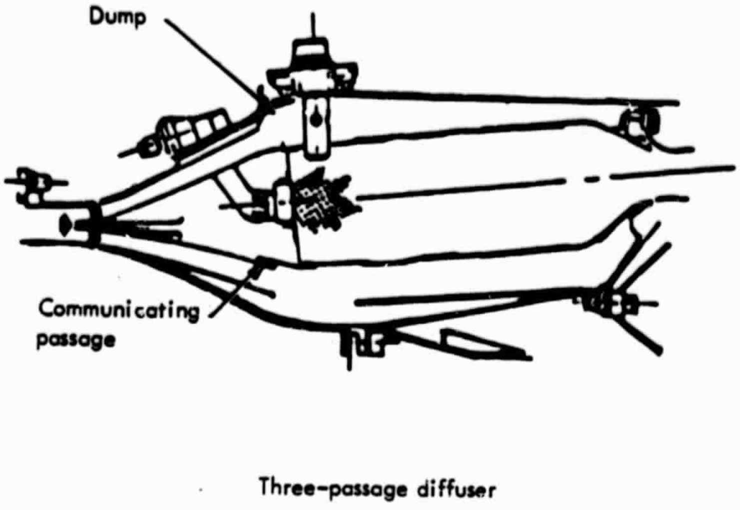
The VCD (shown in Figure 36) is a very short diffuser with 3 major flow paths in lieu of 5 utilized by current 3-passage diffusers. Inner and outer boundary layer control is varied by selecting optimum ratios of inner and outer bleed. The VCD is applicable to the larger STAT engine and offers an improvement in engine performance (0.3% SFC) as a result of a decrease in total pressure drop across the component, as shown in Figure 37. The VCD also requires less axial length (1.9%) than the conventional diffuser, thereby saving engine weight (0.9%) and reducing its acquisition cost (1.7%).

Using the sensitivity values established previously, the following DOC improvements would be achievable in the LCC 50-passenger transport with 3579 kW (4800 hp) engines incorporating a VCD:

	sfc	Cost	Weight	Length	Total DOC
Δ--% (Engine)	-0.3	-1.7	-0.9	-1.9	---
Δ DOC--%	-0.105	-0.068	-0.027	-0.034	-0.234

Lamilloy is a DDA-developed and patented quasitranspiration cooled structural sheet material. It is fabricated by bonding together two or more layers of material that have been etched to form a complex internal flow path as shown on Figure 38. The incorporation of a Lamilloy, two-stage transpiration-cooled combustor in an advanced technology engine provides both weight (2.2%) and price (1.5%) savings.

*Lamilloy is a registered trademark of the General Motors Corporation.
**Numbers in parentheses refer to references listed at the end of this report.



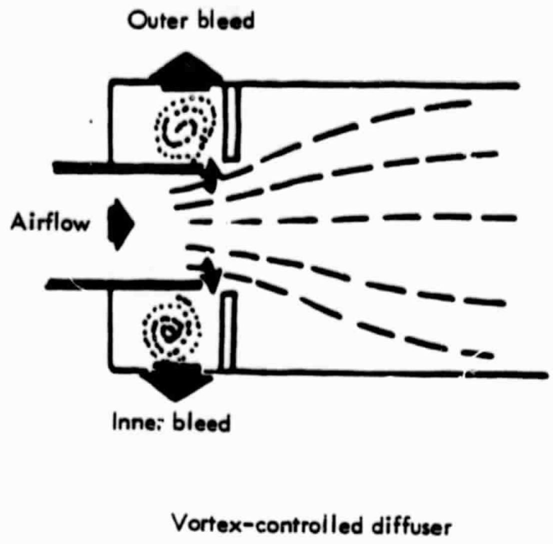


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Figure 36. - DDA diffusers.

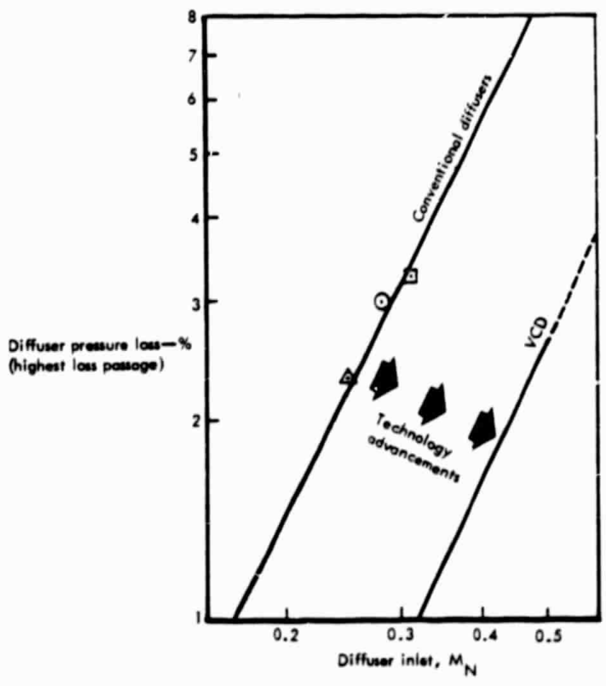

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Figure 37. - Diffuser pressure loss results.

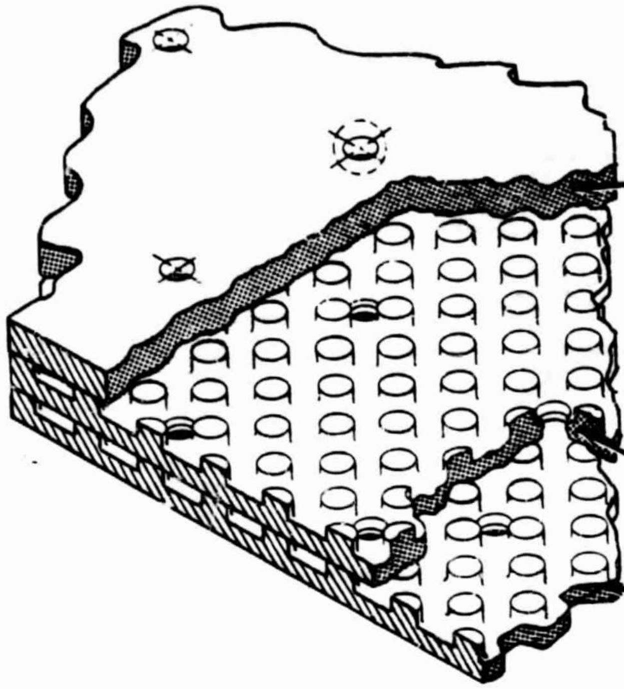


Figure 38. - Transportation-cooled Lamilloy.

Rated power--kW (hp)	% Δ Weight	% Δ Cost	% Δ DOC
1790 (2400)	-2.2	-1.5	-0.224
3579 (4800)	-2.2	-1.5	-0.260

A second-order advantage of the Lamilloy combustor is the reduced requirement for turbine cooling air and improved reliability of the HP turbine as a result of the improved burner out temperature (BOT) profile. Also, the reduced combustor cooling air requirement of Lamilloy makes available more airflow for combustor outlet temperature pattern adjustment.

Accessories

Control Systems

Preliminary conceptual design studies were initiated to identify advanced technology control systems for the STAT advanced technology engines. The criteria for these systems were improved reliability, reduced cost and weight, and improved maintainability as compared to current production engine controls. The control systems would consider total propulsion system requirements to provide for overall thrust management/protection through all required operational conditions.

The control and fuel system will be configured to use an advanced technology digital electronic controller for all required logic and computational requirements for the engine and propeller operation.

The integrated propulsion control system will be designed for high reliability to require less maintenance than current systems, consistent with the requirements of lower TCO and high dispatch reliability. The control system will include the follwoing features:

- o Integration of control functions of engine and propeller to minimize number of systems components
- o Utilization of a full-authority digital electronic controller incorporating advanced, low cost, low power, large scale integration, solid-state components for high reliability for control of both the engine and the propeller
- o Utilization of advanced, simplified fuel pumping and metering components for low cost and weight reduction
- o Self-check capability to detect and provide indication of the occurrence of a malfunction of any of the separate control system components.
- o Optimum location and mounting of the control system components for easy access for routine maintenance and replacement and in a suitable thermal and vibration environment for long life
- o Provisions for remotely actuated devices for all adjustments that may be required in service
- o Provisions for automatic thrust management incorporating ability to select and maintain a number of power control modes for maximum efficiency (takeoff, maximum climb, maximum cruise as a minimum)
- o Provisions of interface with a diagnostic/condition monitoring system sensors for engine health monitoring
- o Provisions for interface, through digital data link, with the airframe flight control systems

Engine Condition Monitoring

The design objective of the STAT propulsion systems will be the achievement of on-condition maintenance whereby scheduled overhauls are eliminated and inspections are minimized. This alone has the potential of eliminating 40% of the current engine, reduction gear, and propeller maintenance cost. A condition that will facilitate the implementation of this maintenance coucept in commercial aircraft service is improved fault detection and isolation via diagnostics to identify impending problems so that corrective action can be taken prior to failure.

To show the advantages of using condition monitoring, an estimation of the DOC and maintenance cost savings for the STAT engines is shown below.

Engine rating--kW (shp)/SLS	1749 (2345)	3544 (4752)
Maintenance cost savings--%Δ	-13.5	-19.2
DOC improvement--%Δ	-1.4	-2.5

A clearly defined on-condition maintenance concept will be developed in conjunction with potential user airlines and the aircraft designers. These concepts will take into account maintenance access times, likely available skill levels, and support equipment. Thus the propulsion system, aircraft, and airline operations can be designed to derive the benefits of condition monitoring equipment. Such equipment can provide an early indication of malfunction and, especially, pinpoint the specific component needing maintenance, thus reducing secondary damage and eliminating shot-gun maintenance of control/accessory components.

Propulsion system condition monitoring provisions will be incorporated in the STAT engines to permit detection of impending malfunctions and to define the required maintenance action. Early detection and correction of potential problems result in improved aircraft safety and reliability. Transducers, which are required to measure component pressures, temperatures, and positions, along with the associated wiring, will be integral parts of the electronic fuel control system.

Advanced technology sensors will be incorporated into the STAT engine condition monitoring systems. These sensors, which are yet to be developed, include those for optical speed and temperature measurement. In addition, advanced low cost compressor discharge pressure sensors and long life gas stream termperature sensors will be incorporated in the STAT control and condition monitoring sytems.

Integration of the control system and engine condition monitoring system into a single electronic system will save engine weight and cost to an extent that could make the following impact on DOC.

Rated power--kW (hp)	% ΔWeight	% ΔCost	% ΔDOC
1790 (2400)	-1.0	-4.0	-.460
3579 (4800)	-1.0	-4.0	-.546

Noise Reduction

The STAT engines with their advanced technology, high pressure ratio cycles will have to be designed so as to minimize their fore and aft noise signatures in order to meet federal regulations and satisfy public demands. If suitable noise suppression technology was not available when the STAT engines were designed, they would have to incorporate inlet and exhaust duct acoustic treatment. With the noise reduction technology available, however, the following impact on DOC might be obtained by removing the duct treatment:

Rated power--kW (hp)	% Δ Weight	% ΔCost	% Δsfc	% Δ DOC
1790 (2400)	-14.0	-1.0	-0.3	-.624
3579 (4800)	-19.0	-1.0	-0.3	-.804

The above analysis gives an insight into the payoff to be realized by developing the technology required to quiet an advance technology engine without resorting to acoustic duct treatment.

Reduction Gear

In considering the various ways to mount a turboprop reduction gear to a power section, the following alternatives are available:

Remote mounted versus integral with engine frame and concentric versus off-set.

There are four possible combinations with the noted alternatives. The choice of the "right" one depends on an assessment of its impact on the following:

- o Weight
- o Complexity--effect on reliability and maintainability
- o Effect on engine inlet; i.e. an integrally mounted concentric gear box could be high in air losses and therefore not be the best choice.
- o Air frame structural requirement; i.e. an offset remote mounted gearbox allows the main wing structure to be aligned with the propeller thrust center line. Another consideration is the desired location of the landing gear. If it were to be in a wing-mounted nacelle, this would impact the desired propulsion system arrangement.
- o Location of, and access to, air frame-required engine driven accessories.

The scope and timing for this STAT study did not permit the full evaluation of the gearbox/power section combinations and their impact on the above factors. It was, therefore, decided to select different arrangements for the two STAT engines which were plausible and would permit full evaluation of both at a later date.

1790 kW (2400 hp) STAT Engine

The 0.5 Mach number flight speed of the Ames aircraft was suitable for a conventional propeller. From the NASA-furnished referenece (2), a design propeller speed of 1250 rpm was obtained. The 1790 kW (2400 shp) engine LP turbine speed of 23,900 rpm established a reduction gear ratio of 19.1:1. A split-torque planetary reduction gear system was well suited for this ratio. The small overall diameter of this system fit well with the integral, concentric design and permitted a short, compact and lightweight engine design. A thorough study of the inlet configuration would be required to provide the desired engine pressure recovery, to avoid a high surface-to-flow-area relationship for the annular scoop, and to improve the level of propeller supercharging.

Aircraft secondary power systems and the engine accessories will be separated to allow engine removal without removal of the aircraft systems. This is accomplished through the use of a remote gearbox with a simple mechanical coupling for the aircraft sytems drives. The propeller control is located on the gearbox forward housing.

3579 kW (4800 hp) STAT Engine

The 0.7 Mach number flight speed of the LCC transport led to the selection of an advanced propfan propulsor (3) for this application. The propfan, with higher disk loadings and rotational speeds than a propeller, had a beneficial impact on the gear train. The power turbine speed of the 3579 kW (4800 shp) engine is 17,760 rpm, which with the propfan speed of 1710 rpm establishes a reduction gear ratio of 10.4:1. This lowered ratio permits the use of a dual-compound idler gearbox. This configuration has fewer gears and bearings and a load-sharing capability to equalize gear and bearing loads. Also, the gear ratio can be altered within a given gear case by simple replacement of the idler set.

This engine incorporates a remote-mounted offset gearbox which provides maximum reliability and installation flexibility. The offset output shaft design permits locating the propfan thrust axis near the wing centerline. Propfan blade tip ground clearance is improved by the offset arrangement. The offset also affords the use of a scoop inlet, which provides higher pressure recoveries than the full annular type.

The propfan controls are located on the aft side of the gearbox facilitating propfan/gearbox/engine integration. A drive is provided for a remote aircraft accessories drive module.

High reliability and low maintenance costs are projected based upon the low number of powertrain gears and bearings, the simplified accessory gear train, and the remote aircraft accessories module.

Composite Materials

DDA designed and produced three composite T56-type reduction gear cases in the late 1960s, and demonstrated the technical feasibility of this approach. The DDA cases were produced by inserting composite preforms with directionally oriented fibers in a closed-form die. The case thus had local stiffening. The most significant problem with this type of material was the excessive cost resulting from the hand labor involved.

Epoxy-based materials have severe temperature limitations (generally 256 K (400°F)). Within this constraint, few engine components could use this technology. The only locations acceptable would be within, or forward of, the compressor inlet housing, and the payoff is a small weight reduction.

Attention has recently turned to composite cases since the cost of fiber has dropped by a factor of 3 since 1970. It has been estimated that the following advantages may be achieved with composite gear cases:

- o Cost: 0 to 10% reduction
- o Weight: 0 to 20% redcution

Based on DDA's T56 composite reduction gear case experience, the most optimistic DOC payoff for a compoaite reduction gearbox appears to be:

	Engine size			
	1790 kW (2400 shp)		3579 kW (4800 shp)	
	Gearbox only	Engine	Gearbox only	Engine
% Δ cost	-2.4	-0.43	-2.4	-0.29
% Δ weight	-4.7	-1.63	-4.7	-1.48
% Δ DOC	---	-0.291	---	-0.20

Steei Gear Teeth With Titanium Web And Shaft

This technology has its payoff in an approximate 20% saving in gear weight. Except for V/STOL aircraft where weight is a significant DOC driver, this technology does not have an appreciable payoff. The gears would be much more expensive and probably show an increase in DOC, if a detailed analysis were possible.

Finite Element Gear Analsys

This technology, or design tool, should reduce redesign effort of gearing and case designs. This is especially true where computer graphics systems supplement the use of three-dimensional finite element techniques. Development of this design tool is a natural extension of systems now in use by industry, and little development should be required. Although engine development effort is slightly reduced by the technology, no appreciable DOC benefit can be identified.

Super-Plastic Formed Titanium Housings

Super-plastic formed titanium housings represent a small potential weight savings and, therefore, would have a marginal DOC benefit to the STAT engines.

Advanced Lubricants

Primary lubricant development efforts are directed toward higher temperature capability. In the STAT commercial engines, higher temperature capability is not an important driver.

Other

Modular Construction

The entire propulsion system will be designed using modular concepts so that failures and resulting removal and repair can be restricted to small equipment packages with little or no disturbance to the rest of the propulsion systems, thus avoiding additional maintenance/shop costs and the opportunity for maintenance errors.

The benefits of modularity include ease of line maintenance, shorter line and shop maintenance repair times, and reduced spare parts requirements. These factors, in turn, reduce aircraft delay times necessitated by component replacement. The price paid for concessions to achieve modularity will, in some cases, include a small weight increase due to incorporation of quick-disconnect features. Another penalty to consider in determining the extent of modularity attainable is the increase in turbine blade tip clearances due to the elimination of line boring major structural members. This was discussed earlier under Clearance Control. Both of these considerations, though not quanitifed during this study, are judged to be more than offset by the advantages of modularity to maintainability.

Remote Accessories

Accessory drives will be isolated and modularized so that the engine or reduction gearbox can be removed without removing most accessories. Also, required maintenance to such modules as accessory drive gearboxes will be performed without removal of the engine or reduction gearbox. The objective will be minimal equipment removal and disturbance, to perform a maintenance action.

Advanced Propellers

Technical evaluation of advanced propeller concepts was accomplished by Hamilton-Standard and McCauley under contract to NASA for future commuter/local service aircraft. Advanced propellers were studied for application at cruise speeds of 0.7 M_N and 0.47 M_N. At 0.7 M_N baseline current technology propellers were referred to the Lockheed Electra application data base using two types of airfoil construction, namely, solid aluminum and the more recent spar-shell technique. Advanced STAT propellers for the 0.7 M_N application featured advanced composite construction, proplets, and precision synchronization with an increase in the number of blades from four to six for the same power input and diameter. The advanced propellers operated at lower tip speed. Cruise effeciency was improved from 78.6% to 85.6% (an improvement of 8.9%), while the fuselage acoustic weight penalty was reduced from 6% of airplane empty weight for a cabin noise level of 98 dB OASPL, to 3.3% of airplane empty weight at a reduced cabin noise level of 85 dB OASPL. Along with these improvements, propeller weight was reduced 28% from the solid aluminum and 3% from the spar-shell current technology propellers. Advanced propeller OEM costs were higher by 17% than the current technology spar-shell type, which is 52% more costly than the solid aluminum type. Evaluating these factors in the DDA 0.7 M_N airplane model results in the improvements for the advanced STAT propeller, compared to the current technology with solid aluminum airfoils, as listed in Table XLI.

TABLE XLI. ATE STAT PROPELLER IMPROVEMENTS FOR THE HIGH SPEED AIRCRAFT

	Base Value	% Change	Empty Weight %Δ	Flyaway Cost %Δ	Block Fuel %Δ	DOC %Δ
Prop fuel average efficiency (Avg. of climb and cruise)	.797	+6	-0.5	-0.4	-6.5	-2.4
Acoustic weight fraction (% of empty weight)	6.0	-45	-3.8	-0.8	-1.5	-0.9
Propeller weight fraction (% of gross weight)	4.6	-28	-2.9	+0.6	-1.1	-0.8
Propeller OEM price fraction (% of aircraft cost)	1.8	+78	---	+2.3	---	+0.4
Net Change*	---	---	-7.2	+0.5	-9.1	-3.7

*Refer to Tables X, XI, and XII for base values of DOC, Block Fuel, Flyaway Cost, and Empty Weight for the high speed airplane.

Thus the advanced STAT propeller for th 0.7 M_N airplane results in net improvements in DOC of 3.7%, in block fuel of 9.1%, and in empty weight of 7.2%, while increasing flyaway cost 0.5%. In addition, cabin noise level was reduced from 98 dB for the current prop to 80 dB OASPL for the advanced STAT prop.

At 0.47 M_N, current technology propellers are the standard general aviation type employing solid aluminum blades with circular shanks. Current technology also include an improved type using lighter weight spar-shell construction and more efficient airfoil shaped shanks. The improved propeller efficiency is 3 percentage points better at cruise than the standard, 15% lighter in weight, but costs about 44% more. The advanced STAT propeller for the 0.47 M_N application (compared to the improved general aviation propeller) features proplets and precision synchronization with an increase in the number of blades from three to six for the same horsepower. Cruise efficiency was improved from 87.5% to 92.3% (an improvement of 5.5%) while the fuselage acoustic weight penalty was reduced from 4.5% of the airplane empty weight to 0%, while holding the 85 dB cabin noise level. These improvements resulted in an increase in weight of the advanced STAT propeller compared to the improved current type, as well as an increase in cost. Weight increased 20.3% and cost increase 155%. Evaluating these factors in the DDA low speed 50 passenger airplane results in changes in DOC, block fuel for the 185.2 km (100 nm) segment, flyaway cost, and empty weight for the advanced STAT 0.47 M_N propeller compared with the improved current type, as listed in Table XLII.

TABLE XLII. ATE STAT P PELLER IMPROVEMENTS FOR THE LOW SPEED AIRCRAFT

	Base Value	% Change	Empty Weight % Δ	Flyaway Cost % Δ	Block Fuel % Δ	DOC % Δ
Prop fuel average efficiency (Avg. of T.O. + 4 x cruise)	0.811	+7.3	-0.8	-0.6	-8.0	-3.1
Acoustic weight fraction (% of empty weight)	4.5	-100	-7.1	-1.7	-3.2	-2.1
Propeller weight fraction (% of gross weight)	2.7	+20.3	+1.3	+0.2	-0.7	+0.3
Propeller OEM price fraction (% of aircraft cost)	1.3	+155	---	+3.2	---	+0.8
Net Change*	---	---	-6.6	+1.1	-10.5	-4.1

*Refer to Tables X, XIII, and XIV for base values of DOC, Block Fuel, Flyaway Cost, and empty weight for low speed airplane.

The advanced STAT 0.47 M_N propeller resulted in significant gains over the improved current technology type when considering the desired cabin noise level of 85 dB OASPL. The new propeller resulted in improvements of 4.1% in DOC, 10.5% in block fuel, and 6.6% in empty weight while increasing acquisition cost 1.1%.

Turbofan Engine

The possibility of using turbofan powerplants for STAT was reviewed. The most likely opportunity was judged to be for the high speed airplane. Combining the performance of the advanced STAT propeller previously discussed with the performance of a high pressure ratio turboprop engine at $0.7M_N$, 6096 m (20,000 ft), maximum continuous power resulted in a thrust sfc of 0.499. This result was compared with the performance of an advanced 6.0 bypass ratio turbofan which had an sfc of 0.631 at the same condition, thus indicating a 21% sfc advantage for the turboprop engine. Based on this indication of the fuel savings potential of the turboprop engine combined with advanced propeller technology, it was decided to concentrate on the turboprop power plant for the purposes of the STAT study.

Bleed and Power Extraction

An assessment was made of the effect of aircraft services provided by the engine on engine performance. Lockheed determined in their STAT studies of 50 passenger short haul aircraft that aircraft requirements amounted to 52.2 kW (70 shp) per engine for electrical, hydraulic, and cabin environmental requirements. A parametric study was conducted using an advanced turboprop engine designed for efficient cruise at a compressor pressure ratio near 20:1

at a rotor inlet temperature of 1366 K (2000°F). The flight condition chosen was 0.7 M_N at 6096m (20,000 ft) altitude. Accessory drive power was extracted from the low pressure spool. Compressor bleed was taken from compressor discharge, and using gas state conditions, an equivalent power was calculated assuming an 80% efficiency for an air driven turbine. Results obtained are shown in Figure 39 for the baseline with no bleed or power extraction, and for cases where the required horsepower was furnished (either all by power extraction or all by bleed), and for the case where half the required horsepower is provided by bleed and half by power extraction. Relative shp is plotted vs relative sfc. Cruise power available to the propeller is reduced 2.1% if 52.2 kW (70 shp) is extracted from the LP spool. This percentage reflects an arithmetical subtraction of the aircraft requirement. For the split case, power loss is 4.6%, and for the all bleed case, power loss is 7.5%. At a given horsepower of 96% of the reference value, relative sfc is 1.008, 1.028, 1.032, and 1.035, respectively, for the baseline, 52.2 kW (70 shp) (shaft), split shaft and bleed, and 52.2 kW (70 shp) (bleed). Thus, for the cases examined, direct power extraction offers the best alternative with respect to minimizing engine performance losses.

Other factors should be evaluated before drawing conclusions on the optimum system to provide aircraft services. These would include a better definition of the power extraction and bleed requirement, the engine power range over which this must be furnished, whether the requirement is continuous or variable, and the trade-offs involved in weight and volume of the environmental conditioning system for variation in air turbine efficiency.

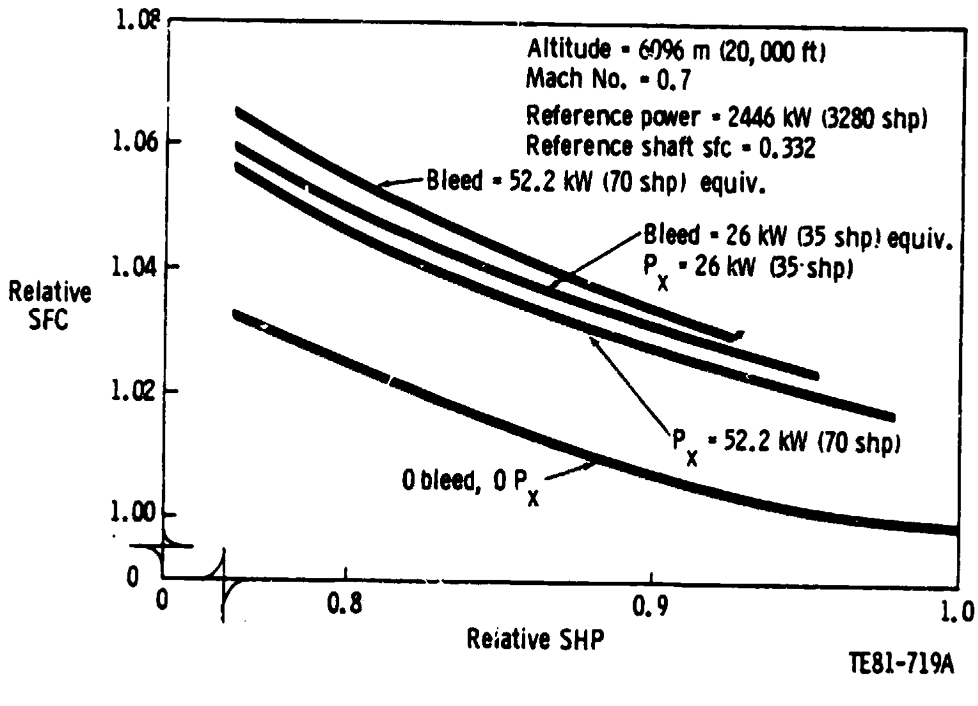


Figure 39. - Impact of compressor bleed and engine power extraction on performance.

Reduced Weight Nacelles

A trade study was conducted to determine the airframe sensitivity to nacelle weight reduction. Figure 40 shows these sensitivities as applicable to both the high speed and low speed aircraft where the nacelle weight fraction approximated 0.2% of the take-off gross weight (actually 0.21% for the high speed aircraft and 0.17% for the low speed aircraft). The lines shown closely approximate the sensitivity of both the high speed and low speed airplanes to change in nacelle weight as it affects empty weight, flyaway cost, block fuel, and DOC.

A significant consideration in the design of the nacelle for the advanced turboprop engine is the inlet configuration, especially for the high speed airplane. The flow field behind the advanced 0.7 M_N STAT propeller should be examined. A screening evaluation of candidate turboprop engine core inlets including concentric types (namely, annular and bifurcated) and scoop types should be undertaken to explore inlet pressure recovery and pressure distortion characteristics.

ADVANCED TECHNOLOGY ENGINES

Two configurations evolved for DDA's STAT advanced technology engines. The basic difference in the engines stems from the two study aircraft power requirements. Engines for the 0.5-Mach Lockheed transport were studied at the nominal size of approximately 1790 kW (2400 shp), and the 0.7-Mach Lockheed transport at 3579 kW (4800 shp). The engines are similar in that both have a single-spool gasifier and a free power turbine with front drive. The overall design and performance features of these engines are shown in Table XLIII and XLIV.

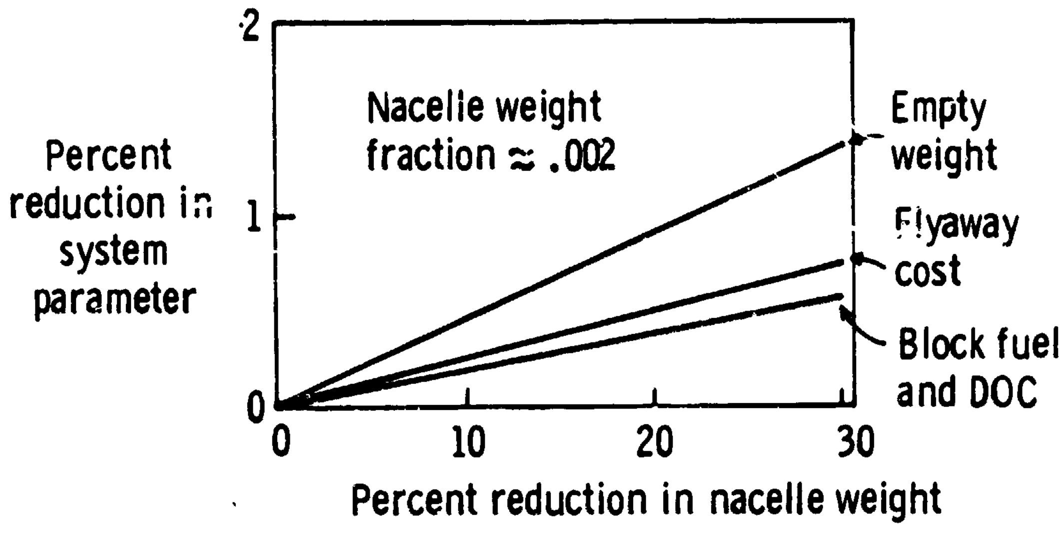


Figure 40. - Sensitivity of aircraft system parameters to nacelle weight.

TABLE XLIII. - ADVANCED STAT TURBOPROP ENGINES (SI UNITS)

	Low Speed Aircraft	High Speed Aircraft
Shaft power--kW	1749	3544
Condition	SLSS	SLSS
Jet thrust--N	1401.19	2491.00
Equivalent shaft power--kW	1843	3711
Cycle		
R_c	20	20
RIT--K	1506	1506
sfc--μg/W·s	71.6	64.5
ESFC--μg/W·s	68.1	61.7
Airflow--kg/s	6.19	11.12
Inlet recovery, %	100	100
Nozzle pressure ratio	1.114	1.114
Gearbox efficiency, %	99	99
Overboard seal leakage, %	1.0	1.0
Technology		
Compressor (type-stages)	Axial-5/cent-1	Axial-9
Turbines (type-stages)	Axial hybrid; HP-2, LP-2	Axial hybrid; HP-2, LP-2
Shafting	Composite	Composite
Components		
Compressors:		
η Adiabatic, %	82.29	83.75
N, rpm	39767	29823
Axial:		
Tip speed ($U_t/\sqrt{\theta}$) m/sec	411.5	457.2
R_c aver./stage	1.35:1	1.4:1
AR blades, aver.	1.2	1.07
Centrifugal:		
N_s (rpm $m^{0.75}/sec^{0.5}$)	28.7	- -
N_s (dimensionless)	0.545	- -
Tip speed ($U_t/\sqrt{\theta}$), m/sec	660.8	- -
R_c	4.47:1	- -
Blade to blade shroud loading, $\overline{LD}$	0.24	- -
Combustors:		
η comb., %	99.9	99.9
$\Delta P/P$ burn.	0.04	0.04
Turbines:		
High pressure:		
η Adiabatic, %	87.78	91.52
Aver. Stage loading coef. ($gJ\Delta h/\bar{U}^2$ mean)	1.60	1.55
Equivalent work ($\Delta h/\theta cr$) MJ/kg	0.102	0.100
Expansion ratio	5.32	4.727
Inlet temperature, K	1505.6	1505.6
Cooling airflow, %	8.3	8.3
Type of cooling (1st blade)	Imp. film, cast-in	Imp. film, cast-in

TABLE XLIII. (CONT)

	Low Speed Aircraft	High Speed Aircraft
Low pressure:		
N, rpm	17,780	23,900
ηAdiabatic, %	89.0	91.2
Aver. stage loading coef.		
($gJ\Delta h/\bar{U}^2$ mean)	1.50	1.68
Equivalent work ($\Delta h/\theta cr$), Btu/lbm	0.076	0.090
Expansion ratio	3.183	3.584
Inlet temperature, K	1063.3	1070.0
Cooling airflow, %	0.5	0.5
Weight--kg		
Power section	189	250
Gearbox (ratio)	130 (19.1:1)	149 (10.4:1)
Total	319	399
Power/weight	5.48	8.88
Length--m		
Power section	1.247	1.285
Overall	1.793	2.215
Max height--m	0.668	0.734
Price*--$		
Power section	243,100	407,600
Total	286,000	468,500
$/kW	164	132

*OEM price at 12/mo, 1000 units, 1979 economics

TABLE XLIV. - ADVANCED STAT TURBOPROP ENGINE (CUSTOMARY UNITS)

	Low Speed Aircraft	High Speed Aircraft
SHP	2345	4752
Condition	SLSS	SLSS
Jet thrust--lb	315	560
ESHP	2471	4976
Cycle		
R_c	20	20
RIT--^oF	2250	2250
sfc--lbm/hp-hr	0.424	0.382
ESFC--lbm/hp-hr	0.403	0.365
Airflow--pps	13.65	24.69
Inlet recovery, %	100	100
Nozzle pressure ratio	1.114	1.114
Gearbox efficiency, %	99	99
Overboard seal leakage, %	1.0	1.0

TABLE XLIV. (CONT)

	Low Speed Aircraft	High Speed Aircraft
Technology		
Compressor (type-stages)	Axial-5/cent.-1	Axial-9
Turbines (type-stages)	Axial, Hybrid; HP-2, LP-2	Axial, Hybrid HP-2, LP-2
Shafting	Composite	Composite
Components		
Compressors:		
η Adiabatic, %	82.29	83.75
N, rpm	39,767	29,823
Axial:		
Tip speed ($U_t/\sqrt{\theta}$), fps	1350	1500
R_c aver./stage	1.35:1	1.4:1
AR blades, aver.	1.20	1.07
Centrifugal:		
N_s, rpm $ft^{0.75}/sec^{0.5}$	70	- -
N_s (dimensionless)	.545	- -
Tip speed ($U_t/\sqrt{\theta}$), fps	2168	- -
R_c	4.47:1	- -
Blade to blade shroud loading, $\overline{LD}$	.24	- -
Combustors:		
η Comb., %	99.9	99.9
$\Delta P/P$ burn.	.04	.04
Turbines:		
High pressure:		
η Adiabatic, %	87.78	91.52
Aver. stage loading coef. ($gJ\Delta h/\overline{U}^2$ mean)	1.60	1.55
Equivalent work ($\Delta h/\theta cr$), Btu/lbm	43.7	42.9
Expansion ratio	5.32	4.727
Inlet temperature, ^{o}R	2710	2710
Cooling airflow, %	8.3	8.3
Type of cooling (first blade)	Imp film, cast-in	Imp film, cast-in
Low pressure:		
N, rpm	17,780	23,900
η Adiabatic, %	89.0	91.2
Aver. stage loading coef. ($gJ\Delta h/\overline{U}^2$ mean)	1.50	1.68
Equivalent work ($\Delta h/\theta cr$), Btu/lbm	32.8	38.8
Expansion ratio	3.183	3.584
Inlet temperature, ^{o}R	1914	1926
Cooling airflow, %	0.5	0.5

TABLE XLIV. (CONT)

	Low Speed Aircraft	High Speed Aircraft
Weight--lb		
Power section	416	551
Gearbox (ratio)	287 (19.1:1)	328 (10.4:1)
Total	703	879
Power/weight	3.34	5.41
Length--in.		
Power section	49.1	50.6
Overall	70.6	87.2
Max height--in.	26.3	28.9
Price*--$		
Power section	243,100	407,600
Total	286,000	468,500
$/SHP	122	99

*OEM price at 12/mo, 1000 units, 1979 economics

1790 kW (2400 shp) STAT Engine

This engine has an axial (5) centrifugal (1) compressor with welded titanium drum and high temperature titanium aft wheels. The final stage of the compressor is a hybrid impeller with cast flow-path ring HIP bonded to a forged hub. The compressor rotor is straddle mounted for dynamic stability and clearance control. The compressor materials will be selected for the rotor, blades, case, and vanes to reduce thermal mismatch and thereby improve clearance control.

The combustor has a fold-back design and features a Lamilloy, transpiration-cooled liner. The two-stage gasifier turbine employs cast-in impingement cooling in both stages plus the first vane stage. Both turbines feature hybrid wheels with cast airfoil rings diffusion bonded to powder alloy hubs. The turbine rotor/case/blades/vanes materials will be selected to minimize thermal growth mismatch. The gasifier turbine incorporates an active clearance control device actuated by the electronic fuel system control.

The turbines incorporate abrasive blade tips and abradable coatings to provide minimum running clearances. Paired knife step seals will be used to reduce seal leakage losses.

Turbine shafts are borsic-titanium matrix composite for light weight and for the required shaft stability in the small diameters required of advance technology, high pressure ratio engines.

The reduction gear for the 1790 kW (2400 shp) engine is close-coupled, coaxially infront of the power section. It incorporates a split-torque planetary gear system with 19.1:1 speed ratio, and features a composite gear case.

C-2

The control and fuel systems for both STAT engines shall be configured to utilize an advanced technology digital electronic controller for all required logic and computational requirements for the engine and propeller operation to accomplish the following control functions:

- o Automatic start sequencing
- o Turbine inlet temperature limiting during all operation, including start, for turbine protection and long life
- o Control for acceleration and deceleration fuel flow, bleed, and compressor geometry for smooth and rapid operation without surge or flameout
- o Control for gas generator speed as a function of power lever input position to provide modulation of engine power from max rating-to-idle-to max reverse
- o Control for propeller/power turbine speed over the required operational range
- o Independent back-up control function that limits maximum power turbine overspeed
- o System for autofeather based upon operating through the propeller control system
- o Provisions for torque limiting for gearbox protection
- o Provisions for automatic mode selections for optimum thrust control (takeoff, maximum climb, maximum cruise as a minimum)
- o Provisions for digital link interfacing with flight control system for automatic propulsion control throughout all regimes of engine operation
- o Propeller synchronizing/synchrophasing

3579 kW (4800 shp) STAT Engine

This engine has a nine-stage axial compressor with welded titanium drum and high-temperature titanium aft wheels. The materials for the rotor, blades, case, and vanes will be selected to reduce thermal mismatch and thereby reduce clearance control.

A VCD forms the transition between the compressor and the flow-through, Lamilloy-cooled annular combustor. The turbine section of this engine features the same number of stages, materials, and cooling technology as that of the 1790 kW (2400 shp) STAT engine.

The reduction gear for the 3579 kW (4800 shp) engine is a remote-mounted, offset type featuring an advanced dual-compound idler gear train with a 10.4:1 reduction ratio. The gear case is made of composite material.

Tables XLV and XLVI show the changes in engine and aircraft parameters for the advanced 1790 kW (2400 shp) and 3579 kW (4800 shp) engines respectively, which incorporate the advanced technologies and design features judged beneficial as described in the previous sections. The total benefit of a group of advance technologies may differ from the sum of the individual benefits attainable from each technology item when considered separately. Therefore, the results presented in Tables XLV and XLVI may differ from the sum of those technology benefits when described individually.

TABLE XLV. - STAT ADVANCED TECHNOLOGY APPLICATION--1749 kW (2400 shp) ENGINE

(ALL VALUES NEGATIVE (-) EXCEPT AS NOTED)

ITEM	$-\Delta\%$ TECHNOLOGY ASSESSMENT (ENGINE PARAMETERS)						$-\Delta\%$ TECHNOLOGY IMPACT SUMMARY (AIRCRAFT PARAMETERS)			
	ACQUIS. COST	MAINT. COST	WEIGHT	PERF. (SFC)	LENGTH	DIAMETER	DOC	FUEL USED	ACQUIS. COST	AIRCRAFT EMPTY WT.
I. Power Section	12.1	45.2	17.0	16.8	6.7	1.0	10.9	19.0	0.5	4.4
A. Improved Modular Design	10.3	38.5	16.6	16.8	6.7	1.0	10.2	19.0	0.5	4.3
o Advanced High R_C Axial-Cent. Compr With Single Shaft Erosion-Resistant Design, IPS, Hybrid Impeller, Clearance Control, And Welded Ti Spool	7.0	27.2	4.3	11.4	2.7	0.5	6.7	12.5	0.3	2.0
o Advanced Turbine With Hybrid Rotors, Composite Shafts, Cast-In Impingement Cooling, Abradable Tip Seals, Clearance Control, And Long Life Bearings.	1.7	10.9	5.6	5.1	4.0	0.5	3.0	5.9	0.1	1.5
o Transpiration-Cooled Combustor	0.6	0	0.9	0	0	0	0.1	0	0	0.1
o Engine Accessories - Electronic Fuel Control, ADV Full Pump And Metering System.	0.5	0.4	0.4	0	0	0	0.1	0	0	0
o Noise Reduction - Compressor And Combustor	0.5	0	5.4	0.3	0	0	0.3	0.6	0.1	0.7
B. Engine Condition Monitoring	1.8	6.7	0.4	0	0	0	0.7	0	0	0.1
II. Reduction Gear	6.8	10.9	+4.4	0	15.1	4.3	1.7	0.6	0.2	0.1
A. Improved Design, Close Coupled, Split-Torque Planetary (19.1:1) With Composite Gear Case And Remote Accessories	6.8	10.6	+4.4	0	15.1	4.3	1.7	0.6	0.2	0.1
B. Engine Condition Monitoring	0	0.3	0	0	0	0	0	0	0	0
III. Turboprop Engine (Totals)	18.9	56.1	12.6	16.8	21.8	5.3	12.6	19.6	0.7	4.5

TABLE XLVI. - STAT ADVANCED TECHNOLOGY APPLICATION--3579 kW (4800 shp) ENGINE

ITEM	-Δ% TECHNOLOGY ASSESSMENT (ENGINE PARAMETERS)						-Δ% TECHNOLOGY IMPACT SUMMARY (AIRCRAFT PARAMETERS)			
	ACQUIS. COST	MAINT. COST	WEIGHT	PERF. (SFC)	LENGTH	DIAMETER	DOC	FUEL USED	ACQUIS. COST	AIRCRAFT EMPTY WT.
I. Power Section	10.3	50.1	24.1	19.0	12.7	4.6	12.8	21.2	0.3	5.3
A. Improved Modular Design	8.7	41.4	23.7	19.0	12.7	4.6	11.8	20.5	0.3	5.2
o Advanced, High R_c Axial Compr. With Single Shaft, Erosion Resistant Design, IPS, Clearance Control, And Welded Ti Spool	4.0	28.4	6.3	13.3	7.0	2.6	7.6	13.7	0.2	2.2
o Advanced Turbine With Hybrid Rotors, Composite Shafts, Cast-In Impingement Cooling, Abradable Tip Seals, Clerance Control, And Long-Life Bearings	2.6	11.5	7.8	5.1	3.8	2.0	3.3	5.7	0.1	1.7
o Transpiration Cooled Combustor With Vortex - Controlled Diffuser	1.3	0	1.2	0.3	1.9	0	0.3	0.4	0	0.2
o Engine Accessories -- Electronic Fuel Control And Adv. Fuel Pump And Metering System	0.4	1.5	0.8	0	0	0	0.2	0.2	0	0.1
o Noise Reduction -- Compressor And Combustor	0.4	0	7.6	0.3	0	0	0.4	0.5	0	1.0
B. Engine Condition Monitoring	1.6	8.7	0.4	0	0	0	1.0	0.7	0	0.1
II. Reduction Gear	6.1	11.9	1.2	0	4.0	8.7	2.4	1.6	0.1	0.7
A. Improved Design, Dual Compound Idler (10.4:1) With Composite Gear Case And Remote Accessories	6.1	11.6	1.2	0	4.0	8.7	2.3	1.6	0.1	0.7
B. Engine Condition Monitoring	0	0.3	0	0	0	0	0.1	0	0	0
III. Turboprop Engine (Totals)	16.4	62.0	25.3	19.0	16.7	13.3	15.2	22.8	0.4	6.0

Maintainability

Propulsion system condition monitoring provisions incorporating advanced technology components will be incorporated in the STAT engines to permit detection of impending malfunctions and to define the required maintenance action. Early detection and correction of potential problems will result in improved aircraft safety and reliability. Transducers which are required to measure component pressures, temperatures and positions, along with the associated wiring, will be integral parts of the electronic fuel control system.

Modular Construction

The entire propulsion system will be designed using modular concepts so that failures and resulting removal and repair will be restricted to small equipment packages, with little or no disturbance to the rest of the propulsion system, thus avoiding excessive maintenance/shop costs and the possibility of maintenance errors.

Engine Noise Considerations

The DDA advanced technology STAT engines, like conventional engines, will radiate noise from the compressor inlet, and combustion, turbine, and jet noise from the engine exhaust. The engine case and reduction gearbox will also radiate noise; however, they and the jet will be minor noise sources. Turbine tones will be substantially above the 10 kHz analysis range (18 kHz at take-off) so that the compressor and combustor remain as the major engine noise sources affecting aircraft far-field noise.

Compressor Noise

The primary impact of the advanced technology will be on compressor noise. The advanced engine cycles that yield low fuel consumption require high pressure ratios, which, in turn, lead directly to supersonic blade tip speeds and increased pressure ratios in the compressor initial stages. The net effect is multiple pure-tone generation and enhanced bladepass tones during take-off and, depending upon the aircraft power requirements, on approach. Engines incorporating advanced technology compressors will require inlet noise suppression in the form of swept first-stage blades, sonic attenuation in the inlet guide vanes, or inlet duct acoustic treatment.

Combustion Noise

Combustion noise levels for the STAT engines are comparatively unknown, since the engine parameters used in the conventional predictions are outside the data base used to generate the prediction empiricisms. In any event, if the STAT engines product 2 to 4 dB more combustion noise than current engines, as predicted, combustion noise will be a major contributor to propulsion system noise. The understanding of combustion noise generation is not sufficiently developed to permit noise reduction either by combustor design or engine cycle bias. Until these controls are developed, combustion noise reduction can be achieved by exhaust duct treatment and/or using shielding.

Propulsion System Noise--Meeting the STAT Goals

Current turboprop aircraft in the STAT-size category just meet or exceed the FAR Part 36 noise requirements, indicating that a propeller noise reduction of 8 dB is required to meet the STAT goal of Part 36-8 EPNdB. A modern, thin, elliptical tip propeller or propfan used in concert with increased take-off performance can provide the required noise reduction. However, unless engine noise (compressor and combustion) is also reduced, the net aircraft gain will be substantially less than the 8 dB required. Figure 41, which is based on measurements of the Electra engine, illustrates this situation. This figure shows clearly that while the propeller of today's technology is the dominant noise source, engine combustion noise provides a floor effect that will, in large measure, negate the propeller propfan noise reduction. The addition of multiple-pure-tones to the engine spectrum will raise the EPNL frequency sensitive portion of the spectrum as well as provide an increase in forward radiation to increase the time during flyover when a given noise level can be perceived on the ground.

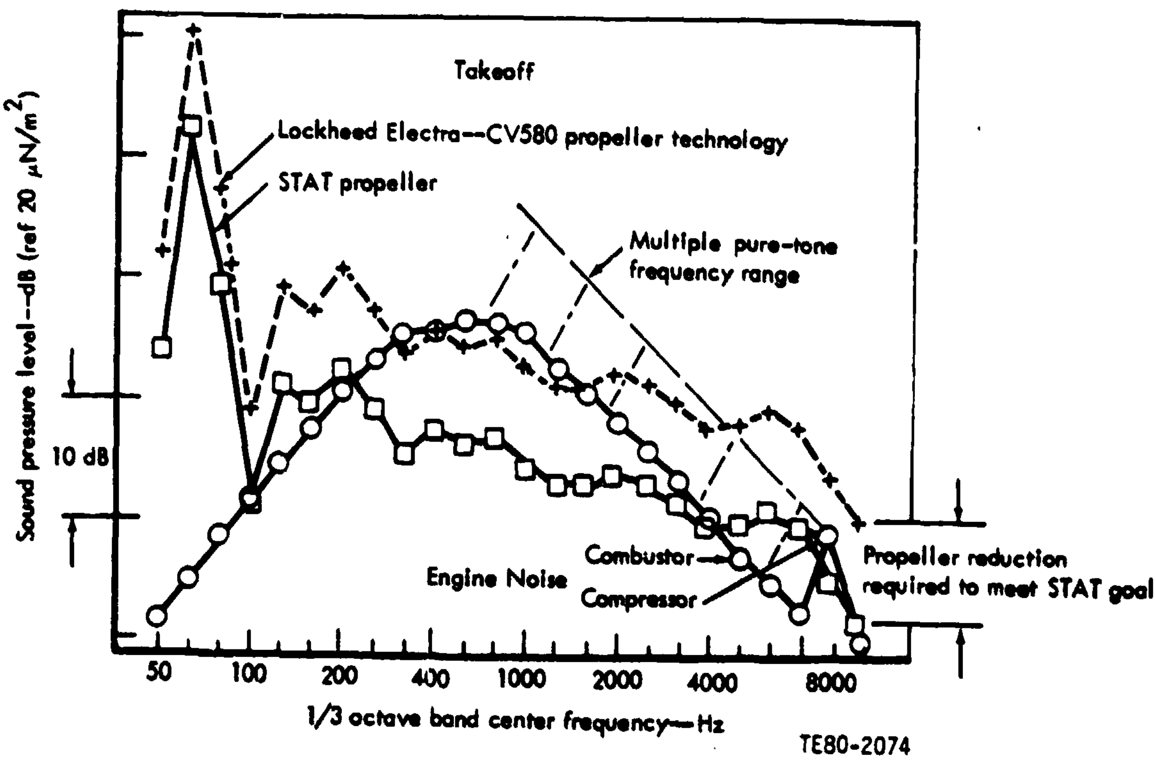


Figure 41. - Comparison of propeller and engine noise during take-off.

In the absence of a well developed noise source reduction technology for turboshaft engines, duct treatment may be used to control engine noise but with an associated increase in fuel consumption, and decrease in power, on the order of 0.5 to 4%. The STAT engines will avoid this performance penalty by giving proper consideration to noise reduction during design and development phases of the components and the engine.

BENEFIT ASSESSMENT

The benefit assessment completes the evaluation of the advanced technology STAT engines by quantifying their effect on the critical airframe and system parameters. In addition to the advanced technology engines, baseline derivative (1985 time frame technology) engines were also analyzed. The resulting data permit comparisons of the potential payoff between:

- o 1990 advanced technology and 1980 baseline technology
- o 1990 advanced technology and 1985 derivative technology
- o 1985 derivative technology and 1980 baseline technology

The evaluation of potential payoff for advanced technology was based on results from a final airframe parameter analysis using the baseline mission described earlier. The airframe and engines were sized to match the required mission, and the resulting operational criteria were quantified. These include:

- o Aircraft DOC for stage lengths of 92.6, 185.2, 277.8, 370.4, 740.8, and 1111.2 km (50, 100, 150, 200, 400, and 600 nm)
- o Fuel used
- o Acquisition price
- o Total cost of ownership for 5 years

based on the following parameters:

- o Fuel cost \$0.264 and \$0.396/L (\$1.00 and \$1.50/gal)
- o 2800 hour/year utilization

Sfc Trends

Sfc trends for current, derivative, and advanced technology turboprop engines are shown in Figure 42 in terms of uninstalled brake sfc as a function of rated shaft horsepower. The trend for current technology turboprop engines resulted from compilations of engine manufacturer's data analyzed by the Lockheed California Company in performance of their STAT Short Haul Study as described earlier in this report. A turboprop version of the XT701 engine, designated PD370-37, established the sfc at the upper end of the horsepower range shown, and other Allison engines such as the Model 501 in the 3729 kW (5000 shp) class and the Model 250 in the 377.9 kW (500 shp) class, influenced the placement of the trend line.

The "derivative" engine technology trend line is based on that for the current technology engines and is keyed to a variant of the PD370-37 in which the compressor pressure ratio is increased from 12.7 to 17.7 and RIT is increased from 1506 to 1533 K (2250 to 2300°F). The derivative engine incorporates a scaled version of a compressor demonstrated in advanced technology programs at DDA and uses the basic shaft, bearing, and turbine arrangements from the XT701 turboshaft engine. The reduction gearbox for the full-scale derivative engine is a new simplified design icorporating a dual-compound idler arrangement and increased design life compared to the T56 series of gearboxes. The new gearbox design is based upon a study of the reliability and maintenance cost history of past turboprop systems and incorporates the recommendations of that study for a gearbox with high reliability, easy maintainability, and low maintenance costs. Realistically, the derivative engine postulated is a major modification involving significant changes in the cold and hot sections of the engine as well as in the gearbox. Performance and maintainability improvements and cost estimates reflect these changes. The sfc improvement is 11% compared to current technology, as shown in Figure 42.

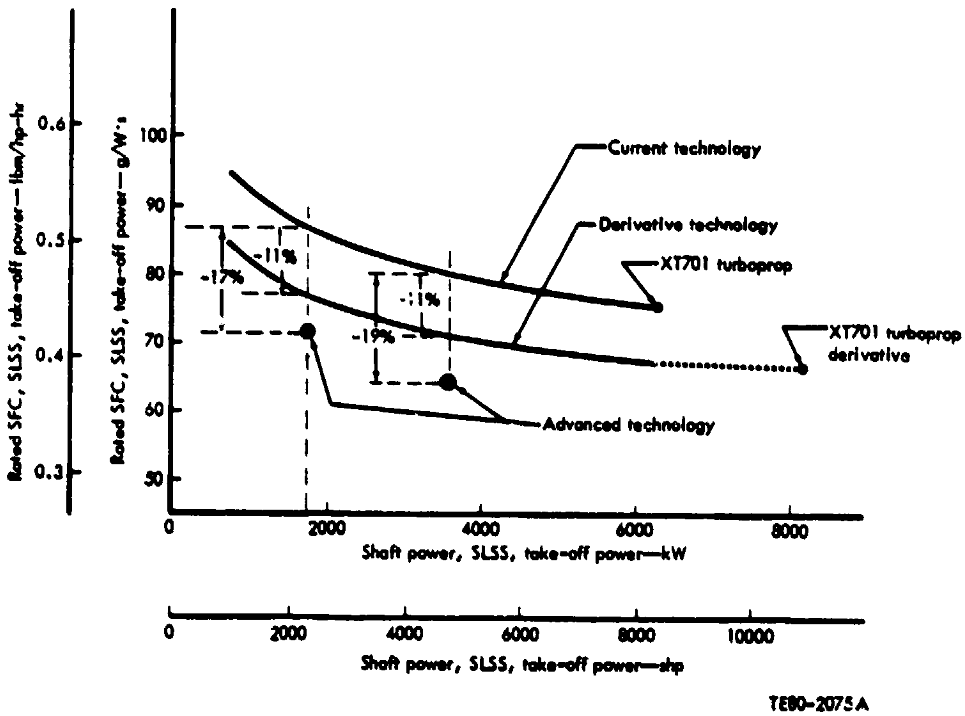


Figure 42. - Sfc trends and comparisons.

The advanced technology engines at 1790 and 3579 kW (2400 and 4800 shp) have an additional improvement in sfc over the derivative engines of 6% and 9% respectively, for a total improvement over current technology of 17% and 19%, respectively.

Price Trends

Historical turboprop production engine prices were reviewed based on DDA data for 3729 kW (5000 shp) and 5966 kW (8000 shp) class engines. The approach was to develop a consistent set of data for extrapolation to STAT sizes. The economic baseline for STAT was:

- o 1979 dollars
- o 1000 engines produced at a rate of 12/month
- o cumulative average price for 1000 engines (with and without inherited learning)

Figure 43 shows relative prices for the T56 and XT701 power sections based on actual DDA experience adjusted to the standards as previously stated, including consideration of the engines as having "no common production base." The learning parameter is considered in order to reflect the price of an available

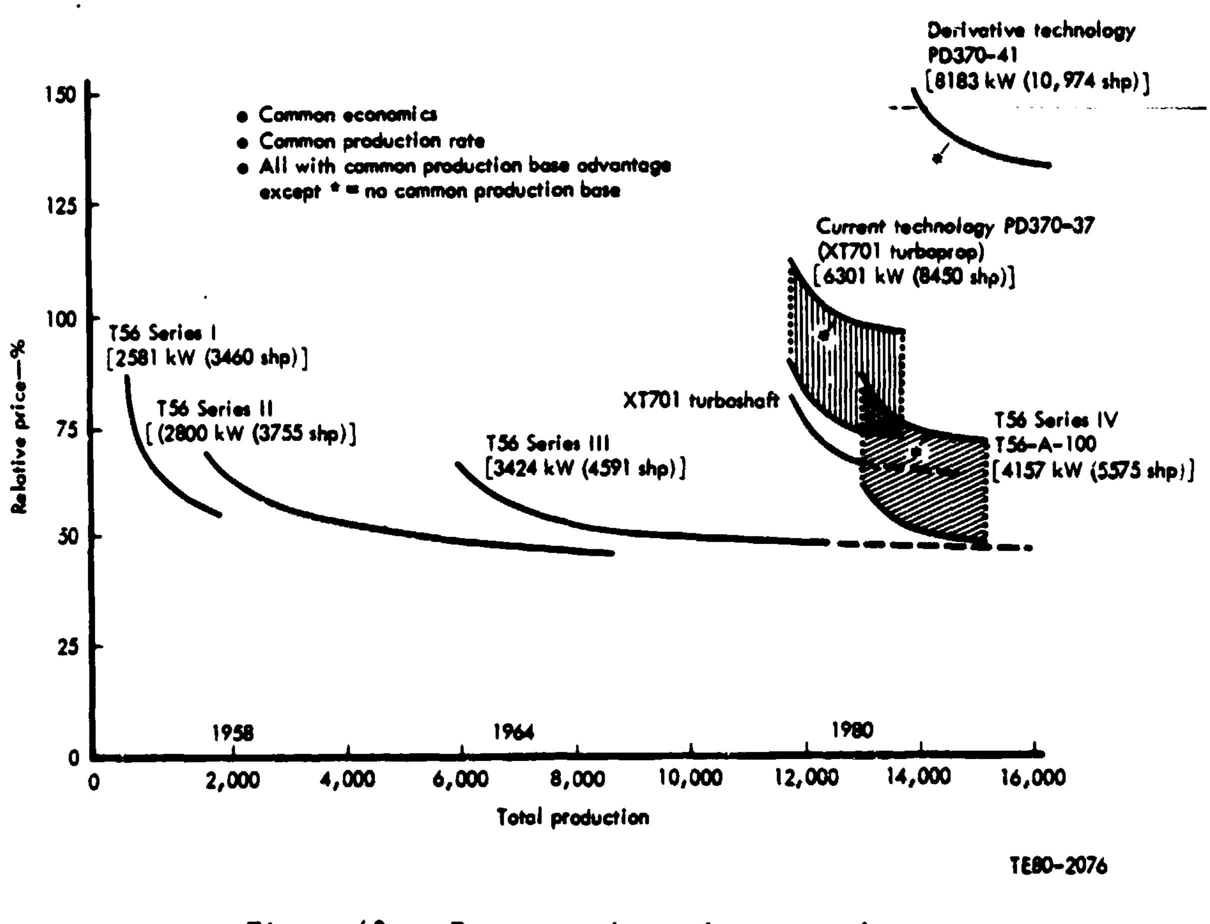


Figure 43. - Power section price comparisons.

current technology engine at the power size required (thus inherited learning applies in pricing) versus a case where a completely new engine is constructed using current technology (thus, in the extreme case, no inherited learning applies).

The model T56-A-100 engine noted in Figure 43 is a current technology Series IV T56 engine with 13,000+ production engines and a 25-year historical background. This engine has redesigned compressor blading to produce a 11.9:1 compressor pressure ratio (CPR) versus the 9.6:1 ratio of the model T56-A-15. The turbine blading has been redesigned to provide 4157 kW (5575 take-off horsepower). The power section weight is 571 kg (1259 lbm). The power sections of the Series I, 2580 kW and 524 kg (3460 shp and 1155 lbm), the Series II, 2800 kW and 540 kg (3755 shp and 1190 lbm), and the Series III, 3424 kW and 553 kg (4591 shp and 1220 lbm) engines all were single spool, axial flow power plants driving an offset reduction gearbox. T56 Series III engines have a 9.6:1 CPR, a 15 kg/s (33 lbm/s) airflow, and a RIT of 1350 K (1970°F).

The projected relative price line shown for the Series IV engine is based on accumulated learning of the T56 production experience. This engine, when factored to "no common production base" is also shown.

The model PD370-37 current technology engine is a turboprop conversion of the XT701 turboshaft engine, where the LP turbine was rematched for better turboprop characteristics. It represents a minimum change from the XT701 engine, which is a free turbine turboshaft engine that was developed through safety demonstration for the U.S. Army's HLH program. The Model PD370-37 is a single-spool, axial flow power unit with a free turbine that is connected by shafting and supporting structure to an offset reduction gear assembly, which is based on an improved T56-A-15 gearbox design. The baseline PD370-37 engine is in the 5966 kW (8000-hp) class. Compressor pressure ratio is 12.7:1, the compressor rpm is 15,049, and RIT is 1506 K (2250°F) for take-off on a standard day. Airflow is 20.8 kg/s (44 lbm/sec), and the power section weight is 510 kg (1125 lbm). Price trends for this engine are also shown with and without inherited learning.

The Model T56-A-100 and the XT701 current technology engines are considered to have the "inherited" learning of approximately 7000 engines, and were adjusted to the 1000 engine basis applying a 90% learning curve consistent with DDA experience.

Derivative engine technology is represented by the Model PD370-41 engine, which is a XT701 turboprop derivative engine with a 17.7:1 compressor pressure ratio. It incorporates a scaled ATEGG demonstrated compressor with combustor and turbine arrangements from the new XT701 turboshaft engine. It is an axial flow engine, having a single-spool core and a front drive free power turbine connected by shafting and supporting structure to an offset reduction gearbox. The engine is in the 8203 kW (11,000 hp) class. At the 17.7:1 pressure ratio at sea level take-off (SLTO), RIT is 1533 K (2300°F), airflow is 24.5 kg/s (54 lbm/sec), compressor rpm is 16,400, power is 8183 kW (10,974 shp), and the power section weight is 682 kg (1503 lbm). This engine has an advanced design gearbox of the dual-compound idler concept, which is lighter in weight than the current technology gearbox.

Data from Figure 43 is used to establish power section OEM comparative prices without inherited learning, shown in Table XLVII. Gearbox price estimates are added to the power section price to obtain a total engine price comparison without inherited learning. The gearbox costs have been factored from historical costs similar to the power section. However, a 94% learning curve was applied for current technology gearboxes.

TABLE XLVII. - ENGINE PRICE SUMMARY

Engine base	T56, Series IV (Model T56-A-100)	XT701 (Model PD370-37)	XT701 derivative (Model PD370-41)
Technology	Current	Current	Derivative
Size--kW (shp)	4157 (5575)	6301 (8450)	8184 (10975)
*Power section price	$582,200	$782,800	$1,079,200
*Reduction gear price	$102,400	$114,900	$88,000
*Engine price	$684,600	$897,700	$1,167,200

*Cumulative average OEM prices of first 1000 engines at 12/mo, 1979 economics, without learning or benefit of previous production base.

Price trends taken from the data points shown in Table XLVII were then constructed (using the scaling relationship: scaled price = unity price x [scaled power/unity power]$^{0.7}$), and are shown in Figure 44. Note that the T56-A-100 falls very close to the current technology specific price versus take-off rated shp "trend line." Also note that the derivative engine trend curve parallels the current technology trend line, and is 5 to 7% higher than the current technology lines as a result of the new compressor and turbine design and the increase of RIT to 1533 K (2300°F).

The advanced STAT technology 3579 kW (4800 shp) engine is a high pressure ratio conceptual engine with a compressor pressure ratio of 20:1, a W_a of 11.6 kg/s (25.5 lbm/sec), an RIT of 1506 K (2250°F), and a 29,800 compressor rpm. The baseline engine power section has been calculated to weigh 250 kg (551 lbm). The engine is a single-spool, axial-flow power unit with a free turbine connected to an offset reduction gear assembly.

This STAT engine is approximately 16% lower in cost than current technology engines of the same power. The small compressor diameter and overall low weight of the power section plus the new design gearbox contribute largely to this low cost. The costs were derived parametrically from known costs on DDA production and demonstrator engines and DDA MIF methodology and have been factored to a common basis.

The advanced STAT technology 1790 kW (2500 shp) engine has an axial compressor combined with a single-stage centrifugal outlet stage. This engine has a 20:1 CPR, 6.35-kg/s (14.0-lbm/sec) airflow, 38,700 rpm, and a 1506 K (2250°F) RIT,

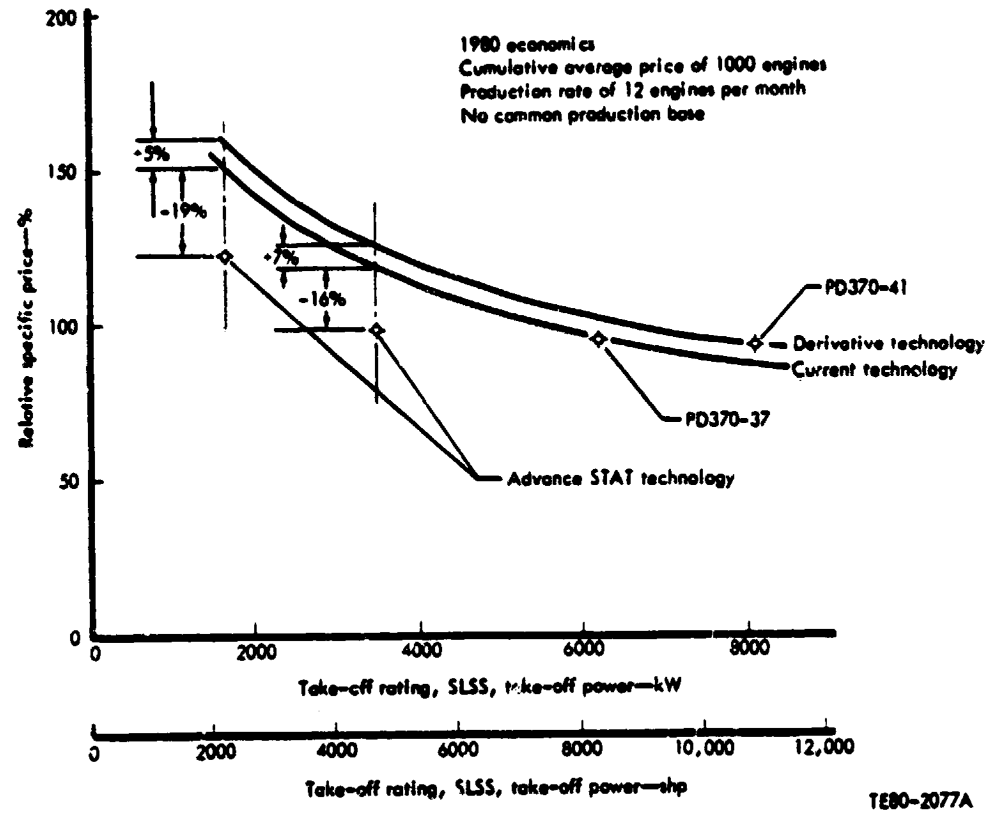


Figure 44. - Engine specific price comparisons.

with a power section weight of 188.7 kg (416 lbm). This STAT engine is a single-spool, axial-centrifugal power unit with a free turbine connected to an in-line reduction gear assembly of advance design.

Maintenance Cost Trends

Advanced Turboprop Maintenance and Reliability Recommendations

A study of actual turboprop reliability and maintenance costs in commercial operation, conducted by DDA for NASA and reported in NASA CR 135192, showed that turboprop maintenance costs could be reduced in future propulsion systems by incorporation or improvement of the following elements:

- o Incorporation of on-condition maintenance
- o Improved modularity
- o Incorporation of total system management to integrate all elements of the propulsion system.
- o Improved reliability and durability

On-Condition Maintenance

A prime cost driver in the maintenance of aircraft propulsion systems can be scheduled overhauls. It was found in the reliability and maintainability study for NASA that scheduled overhauls accounted for 40% of the maintenance costs of the turboprops on the Lockheed Electra and Convair CV-580s. A potential for most of the 40% reduction may, therefore, be realized for the STAT engines by adopting on-condition maintenance. Scheduled overhauls are eliminated, and periodic inspection or equipment malfunction are the only sources to justify equipment removal. The on-condition maintenance concept can be facilitated by improved fault detection and isolation via diagnostics (condition monitoring) to identify impending problems so that corrective action may be taken prior to failure. Most of the large turbofan engines currently in use with the aircraft of the major airlines are maintained on-condition, although some section of a given engine, such as the high pressure turbine, may have a time-limited removal requirement based upon operating experience. All-up condition monitoring systems are still in development and have not been adopted by the major airlines, but they should be operational by the 1990s. The estimated reduction in maintenance costs for the adoption of on-condition maintenance and condition monitoring is discussed later in this section.

Total System Management

The management of the total propulsion system must be centralized, including the propeller, engine, and nacelle. This would ensure the proper definition and control of interfaces between the three major system modules, which would minimize independent approaches to major module design. Maintenance access to all components and total system condition monitoring would be well integrated.

Reliability and Life Goals

The reliability and life goals for the advanced STAT engines were established consistent with safety, a minimum maintenace cost for the mature system, minimum delays and down time, and the expected service life of the engines in short haul commuter airline service.

Life Goals

Design life was defined as the time or life that the propulsion sytem would operate satisfactorily:

- With routine scheduled maintenance
- Without scheduled replacement of parts or components
- With unscheduled replacment frequencies no more than are consistent with the stated MTBF (mean time between failure) values

Using the guidelines of the contract work statement and other direct operating ground rules supplied by NASA, in which airframe design life of at least 30,000 hours, 60,000 take-off and landing cycles, and a spares factor of 1.3 for engines were defined as requirements, the following life goals for the engines were established:

- Design life of high pressure turbine of 7500 hours
- Design life of all other parts of 25,000 hours

Reliability Goals

Reliability goals were established for the power sections and gearboxes of the STAT engines. These goals were based on the incorporation of all advance technology items and are considerably improved over those of current technology engines. Mean time between unscheduled removal (MTBR), based upon propulsion system inherent events, should be no less than those shown in Table XLVIII for major modules, and Table XLIX for components or accessories. For comparison, this table also shows removal rates for nearly 2.5 million flight-hours of experience on the Series I T56 engines.

TABLE XLVIII. - INHERENT RELIABILITY GOALS FOR STAT ADVANCED TURBOPROP - MAJOR MODULES

Major module	Inherent MTBR--h	Corresponding removal rate/1000 h	Ref. T56, Series I removel rate/1000 h
Core engine	6,250	0.160	0.426
LP (power) turbine	50,000	0.020	
Main drive reduction gearbox	33,333	0.030	0.060
Total for major modules	4,762	0.210	0.486

TABLE XLIX. - INHERENT RELIABILITY GOALS FOR STAT ADVANCED TURBOPROP--COMPONENTS AND ACCESSORIES

Components and accessories	Inherent MTBR--h	Corresponding removal rate/1000 h
Power section major accessories (oil, scavenge, fuel pumps, ignition)	50,000	0.020
Engine accessory drive gearbox	50,000	0.020
Power section minor accessories	6,667	0.150
Control System	2,500	0.400
Start system	3,700	0.270
Total for major components	1,163	0.860

Maintenance Philosophy

Maintenance philosophy is defined as a characteristic of equipment design that facilitates maximum maintenance effectiveness and minimum TCO. Cost-effective maintenace concepts must be designed into the end product during the conceptual and design phases. An on-condition maintenance approach, as described earlier in this section, is a result of this philosphy and is facilitated by the improved durability, condition monitoring with improved diagnostics, increased modularity, and simplified hardware. Other specific design requirments for improved maintainability are methods of mounting for easy installation and removal, accessibility, repairability, inspection/verification concepts, serviceability, and component detail design for ease of maintenance.

Reliability Assessment

A reliability assessment was made of the STAT advanced turboprop engines to provide the predicted improvements, in contrast to current systems and to provide a basis for maintenance cost predictions.

Inherent Versus Noninherent Reliability

Inherent events/removals are those caused primarily be propulsion system equipment failures. Noninherent reliability is based upon those events that are primarily not caused by propulsion system equipment. In order to make direct comparison with current technology rate and cost data, which reflect total or "operational" removal rates and cost, an allowance had to be provided for noninherent events. This allowance has the effect of increasing the removal rates beyond the inherent value or decreasing the MTBR values below the inherent value. The noninherent allowance provided for the effects of the following:

- o Unsubstantiated/unnecessary removals
- o Improper maintenance-caused failures
- o Foreign object damage (FOD)
- o Convenience to perform nonpropulsion maintenance
- o Accident damage

Summary of Total Operational Reliability Assessment

The total operational reliability assessment is shown in Tables L and LI. These values were used to determine maintenance costs. Also shown are the values of the inherent and noninherent rates used to arrive at the predicted operational rates. The inherent and noninherent reliability assessments are discussed in the following subsections.

TABLE L. - SUMMARY OF RELIABILITY ASSESSMENTS OF STAT ADVANCED TURBOPROP SYSTEM--MAJOR MODULES

Name of major module	Inherent removal rate/1000 h	Noninherent removal rate/1000 h	Operational removal rate/1000 h	Operational MTBR--h
Core engine	0.160	0.040	0.200	5,000
LP turbine	0.020	---	0.020	50,000
Main drive reduction gearbox	0.030	0.010	0.040	25,000
Total for major modules	0.210	0.050	0.260	3,850

TABLE LI. - SUMMARY OF RELIABILITY ASSESSMENTS OF STAT ADVANCED SYSTEM--COMPONENTS AND ACCESSORIES

Name of component or accessory	Inherent removal rate/1000 h	Noninherent removal rate/1000 h	Operational removal rate/1000 h	Operational MTBR--h
Engine accessory gearbox	0.020	0.005	0.025	40,000
Start system	0.200	0.133	0.333	3,000
Control system	0.400	0.100	0.500	2,000
Major accessories (oil pumps, fuel pumps, ignition)	0.020	0.016	0.036	27,777
Minor accessories	0.150	0.050	0.200	5,000
Total for components and accessories	0.790	0.304	1.094	914

Core Engine and LP Turbine

Inherent Reliability Assessment

The history of DDA turboprop experience was studied from two important aspects:

- o Whether reliability problems existed that were uniquely related to turboprop operation or application
- o Identification of principal engine problems to evaluate solutions from current technology

No current engine problems were identified as being unique to turboprop operations or application. Some of the problems resulted from the engines being originally designed and developed for much shorter life in military use and without the comprehensive design criteria of today. Thus, one aspect of the core engine and LP turbine assessment recognizes the beneficial results of comprehensive design criteria, which include clearly stated reliability and life requiremets for commercial operation.

Some of the corrective actions to historical problem areas are straightforward configuration and processing changes to better adapt to commercial maintenance plans and commercial overhaul periods. Others were based on improvements from technology and analytical improvement programs. It is assumed that this trend will continue to produce improved materials, coatings, and analytical techniques available for the advanced STAT engines.

Based upon the reliability of current mature engines, the projected inherent reliability of the advanced core is 6250 hours MTBR and of the LP turbine, 50,000-hour MTBR. The corresponding inherent PRR are 0.16/1000 and 0.02/1000 hours, respectively.

Noninherent Reliability Assessment

An assessment of 0.040 removals for 1000 hours was made for the core engine. The causes of these would be expected to be FOD to the compressor, improper maintenance and operation, and unsubstantiation (no fault found). With fault diagnostic systems employed and the beneficial effect on improper maintenace resulting from a lower maintenance action rate, the estimate of 0.040 seems reasonable.

Main Drive Reduction Gearbox

Inherent Reliability Assessment

The assessment of the STAT advanced main drive reduction gearboxes, each a major module, was based upon current engine experience. For example, the study of turboprop reliability and maintenance costs for NASA, which were mentioned earlier in this section, found that 36% of the reduction gearbox removals were caused by failures in the main drive system or in other characteristic turboprop functions such as propeller brake, safety coupling, and negative torque signal (NTS). The advanced main drive reduction gearboxes contain those functions performed by the 36% represented previously, less the safety coupling and NTS, plus the limited accessory drive gears, bearings, and associated hardware necessary to drive the lube pump, propeller high pressure pump, and the power take-off for the remote aircraft accessory gearbox. Therefore, the advanced main drive reduction gear system can be compared directly with the corresponding portion of the current gearboxes. In the study, it was found that the main drive system (the 36%) accounted for:

- An inherent premature removal rate of 0.048 per 1000 engine flight hours (EFH) (equivalent MTBR = 20,770 hours)
- A total operational premature removal rate of 0.061 per 1000 EFH (equivalent MTBR = 16,400 hours)

The advanced main drive reduction gear systems:

- Contain less than half the number of powertrain bearings of the current main drive system
- Contain proportionately fewer other hardware parts
- Must be expected to operate for 25,000 hours with no scheduled overhaul (compared to about 7000 hours average for the current system)

The first two points produce a favorable effect on premature removal rates, whereas the third tends to produce an unfavorable effect. To control and minimize this potential unfavorable effect, one feature that the advanced designs will incorporate is bearing sizes selected for much longer life. Some further reduction in the failure rate is expected as a result of the improved operating conditions afforded by use of helical gears and improved mounting and gear support.

Improved alignment, concentricity, and reduction of torsionals in the remaining accessory drive train will contribute to attainment of full calculated life at these locations. These factors, however, cannot be easily quantified to produce a further improved failure rate value.

Considering all these factors and conducting a reliability analytical analysis with the record of the current system as a baseline, a premature removal rate of 0.01 per 1000 hours was determined for the advanced main drive systems.

A premature removal rate of 0.01 per 1000 hours was assumed for the accessory drive system within the main drive reduction gearbox. This assumption was based upon the complexity of the accessory drive system being comparable to that of the main drive reduction system and that the same reliability design criteria would be used for the accessory drive as for the main drive.

More frequent internal inspection opportunities exist for the current technology reduction gearboxes than forecasted for the advanced main drive reduction gearboxes. Current technology reduction gearboxes are disassembled for repair and inspection after each premature, and TBO, removal. During the inspection performed at the repairs and overhauls, distressed parts are replaced. With an "on-condition" operation and the simplified, more reliable gearbox there will not be the frequent opportunity for replacement of parts. The effect may be a slight increase in the premature removal rate (PRR) that was predicted solely from the improvement in the designs. An additional rate of 0.01 per 1000 hours was assumed for this effect.

The inherent reliability predictions for the STAT advanced main drive reduction gearboxes are summarized in Table LII.

TABLE LII. - INHERENT RELIABILITY PREDICTION SUMMARY FOR STAT ADVANCED MAIN DRIVE REDUCTION GEARBOX.

Gearbox system	Inherent premature removal rate per 1000 h
Main reduction system	0.010
Accessory drives for two pumps and remote gearbox	0.010
Effect of less frequent disassembly inspections	0.010
Total predicted for main drive gearbox	0.030
Equivalent MTBR	33,000 hours

Noninherent Reliability Assessments

The principal noninherent removals of the main drive reduction gearbox are for:

- o Improper maintenance
- o Unsubstantiated (no failure found)
- o Accident damage

Accident damage to the gearbox results from unusually large forces transferred to the gearbox from the propeller in case of an accident. The rate of accident damage removals plus precautionary removals after an accident are a function of the propeller accident rate. Projected improved propeller accident rates supplied by Hamilton Standard were used to estimate noninherent removals of the gearbox for this cause.

The noninherent removal rates for improper maintenance and unsubstantiated causes were estimated from studying the historical data base and the relative complexity of current technology and advanced gearboxes.

The estimated noninherent rate for the STAT advanced main drive gearboses is one-third of the inherent rate, or 0.010/1000 hours.

Power Section Accessory Gearbox, Components, and Accessories

Inherent Reliability Assessment

The basis for the reliability assessments of power section accessories were detailed studies made for the XT701 engine (most in conjunction with the suppliers), listing of the Allison Model 501 including the more recent Series III (Model 501-D22 and the T56-A-14 and -15), and DDA/supplier estimates for electronic control systems.

Noninherent Reliability Assessment

Estimates were made for component and accessory removal rates for noninherent causes using the same background data, studies, and supplier experience as discussed for the inherent reliability assessment.

Maintenance Cost Projections

Baseline Engine

The determination of the baseline maintenance cost of current technology engines and reduction gearboxes was described previously in the Baseline Engines section, and the trend with horsepower is shown in Figure 45. This trend was established by Lockheed using cost data and scaling characteristic supplied by DDA, General Electric, and Garrett for current turboprop engines.

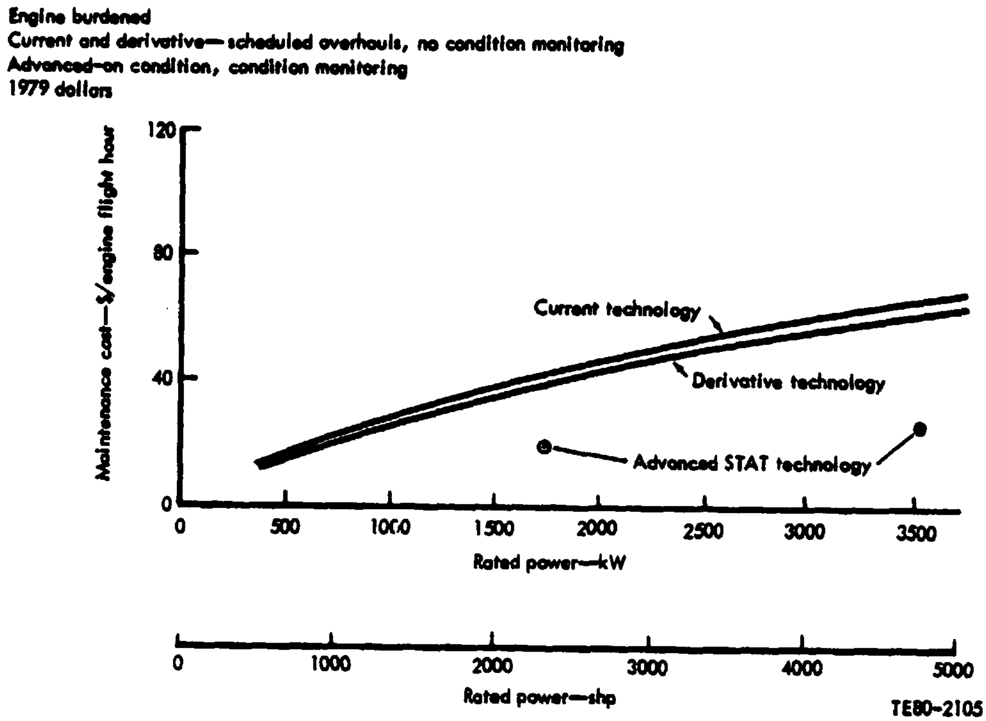


Figure 45. - Engine maintenance cost versus rated power.

Derivative Engine

DDA derivative engine technology was defined as that equivalent to a derivative of the Model PD370-37 engine and designated as the Model PD 370-41 engine in the Navy MPA studies. Scaled characteristics of this derivative engine were derived for use in this STAT study.

The derivative engine differed from the baseline engine by having a booster stage added to the compressor and a new main drive reduction gearbox that was based upon the results of the study of turboprop reliability and maintenance costs. The added booster stage increased the pressure ratio from 12.5:1 to 17.7:1.

The following assumptions were made in establishing the maintenance costs in comparison to the current technology engine:

- o The main drive reduction gearbox would be designed to the "on-condition maintenace philosophy with no scheduled overhauls
- o A condition monitoring system would not be available on the derivative engine
- o Improvements in the core engine would increase mean time between overhaul (MTBO) by 8.5%

The maintenance costs of the 1749- and 3544-kW (2345 and 4752 shp) derivative engines were estimated to be 8.4% lower than the current technology engines. A comparison is shown in Table LIII.

TABLE LIII. - MAINTANANCE COST COMPARISON FOR BASELINE AND DERIVATIVE ENGINES

Engine rating--kW (shp)/SLS	1749 (2345)	3544 (4752)
Maintanance cost--$/EFH		
Baseline engine	40.73	64.91
Derivative engine	37.38	59.57

The reduction in costs attributed to the derivative engines is primarily the result of the improved main drive reduction gearbox. The gearbox reduced the cost by 9.2%, while the power section increased in cost by 0.8%. The increased cost for the power section resulted from the net effect of increased parts cost and decreased overhaul removal rate.

Advanced STAT Engines

The maintenance cost projections for the advanced STAT engines were estimated by multiplying the line and shop labor and material charges per maintenance action by the corresponding rate of maintenance action, or repairs. Labor costs expressed in 1979 dollars were based on a direct labor rate of $10.00/h and a burden labor rate of $18.00/h. Material costs per repair were developed using estimated acquisition costs and historical data relating per-repair material costs to acquisition costs on a percentage basis.

The results of the maintenance cost projection is shown in Table LIV and Figure 45, where they are also compared with the baseline (current technology) and derivative engines.

TABLE LIV. - MAINTENANCE COST PROJECTION OF ADVANCED STAT ENGINES COMPARED WITH BASELINE AND DERIVATIVE ENGINES

Engine rating--kW (shp)/SLS	1749 (2345)	3544 (4752)
Maintenance cost--$/EFH		
Baseline engines	40.73	64.91
Derivative engines	37.38	59.57
Advanced engines	17.87	24.64

The improvement in maintenance cost of the derivative engines over the baseline engines is due primarily to incorporation of advanced reduction gearboxes. The further improvement of the advance engines is due to the incorporation of advance technologies which improve reliability and facilitate maintenance actions. The cost projections of advanced engine maintenance over that of the baseline engine, are such that the 1749 kW (2345 shp) engine cost will be reduced by about 56% and the 3544 kW (4752 shp) engine cost will be reduced by about 62%. These reductions are the net effect of the design, reliability, and maintainability features described previously.

To show the advantages of using condition monitoring, an estimation of the maintenance costs without the use of condition monitoring was also made. Comparison with and without condition monitoring are shown in Table LV.

TABLE LV. - MAINTENANCE COST PROJECTIONS OF ADVANCED STAT ENGINES WITH AND WITHOUT CONDITION MONITORING

Engine rating--kW (shp)/SLS	1749 (2345)	3544 (4752)
Maintenance cost--$/EFH		
With condition monitoring	17.87	24.64
Without condition monitoring	20.67	30.50

These costs were estimated by adjusting the rate of removals and the increased damage (material costs) that would occur before detection of a failure could be made. Instead of the 56% for the 1749 kW (2345 shp) and 62% percent for the 3544 kW (4752 shp) maintenance cost savings, they would be reduced to about 49 and 53%, respectively. Thus, the net effect of condition monitoring is about 7 to 9% in cost savings compared to the baseline. Comparing a given engine with and without condition monitoring, this feature reduces maintanance costs by about 14% for the 1749 kW (2345 shp) to 19% for the 3544 kW (4752 shp) engines.

To show the effect of on-condition maintanance, an estimation of the maintenance costs of the 3544 kW (4752 shp) engine was made by assuming scheduled overhauls of 10,000 h on the cold section and 5000 h on the hot section. Comparisons are shown in Table LVI.

TABLE LVI. - MAINTENANCE COST PROJECTION OF 3544 kW (4752 shp) ADVANCED STAT ENGINE WITH AND WITHOUT ON-CONDITION

	On-condition--7500-h TBO-HP turbine	Scheduled overhauls--10,000-h cold section & 5,000-h hot section
Maintenance cost--$/EFH		
With condition monitoring	24.64	39.82
Without condition monitoring	30.50	46.21

These estimations were made by adjusting the rate of unscheduled removals downward and inserting the rate for scheduled removals. Material and labor costs per action were also adjusted to account for the effect of increased inspection during scheduled overhauls on the extent of damage at failure. Scheduled maintenance (overhauls) increased the cost about 62% with condition monitoring and 52% without it. Compared to the baseline engine, without condition monitoring, the cost reduction of the advanced engine with scheduled overhauls is about 29%.

This discussion shows that maintenance cost improvements from the baseline engine can vary from about 29 to 62%, depending upon whether the maintenance concepts of on-condition and condition monitoring are used. The 29% improvement for the Advance STAT engines results from the incorporation of advance technologies which improve the reliability of the engines. Refer to Tables XLV and XLVI for a breakdown of these maintenance costs.

Weight Trends

Specific weight trends estimated for the STAT engines, as compared to current and derivative technology trends, are shown in Figure 46. The current technology line was based on results from Lockheed's studies for NASA Ames. Derivative technology was constructed to parallel the current technology trend line and was keyed to the weight of the XT701 turboprop derivative engine discussed earlier. The 1790 kW (2400 shp) STAT engine was 14% better in specific weight than the current technology reference. The 3579 kW (4800 shp) engine was 34% better. The 3579 kW (4800 shp) engine has greater improvement primarily because the reduction gear ratio is much lower as the result of driving a propfan rather than a conventional propeller.

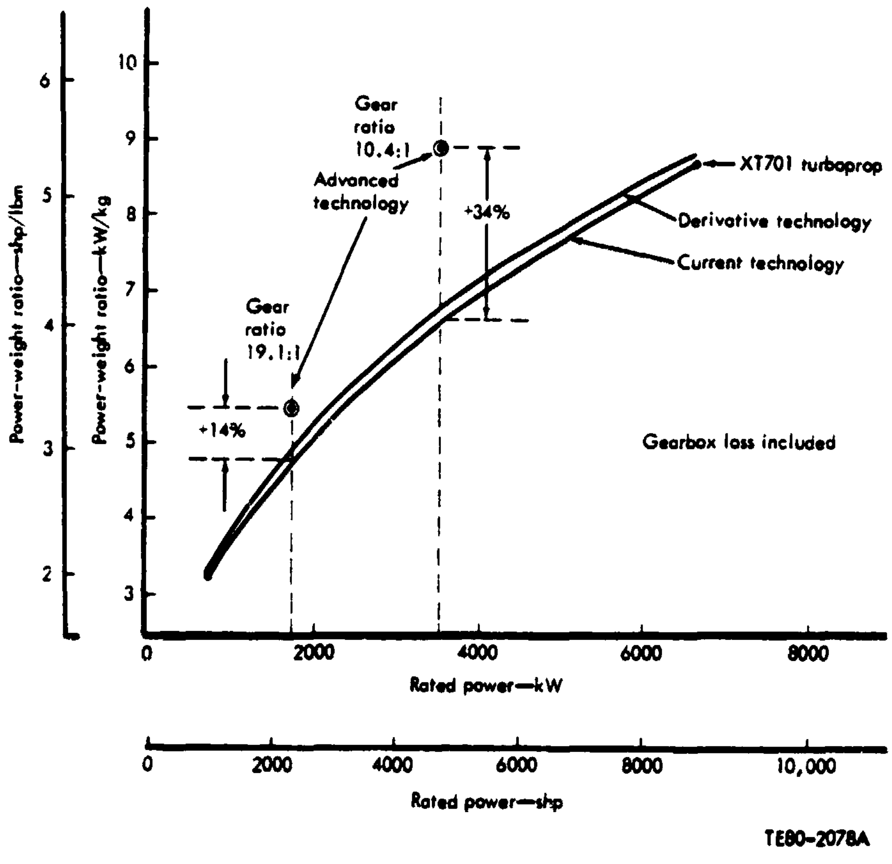


Figure 46. - STAT specific weight trends.

Engine Technology Comparisons

Current (CTE), derivative (DTE), and advanced (ATE) STAT engine comparisons in terms of sfc, weight, price, maintenance cost, and maximum envelope length and height at the rated condition are shown in Tables LVII and LVIII. (Note that the current technology engine 'CTE' is the same as CTE, except that the 'CTE' denotes "no inherited learning".) Engine prices, at STAT baseline economics, are shown on a common basis with respect to learning, i.e., without benefit of common production base. Maintenance costs are for a mature engine. Note that the STAT engines achieve significant improvements in sfc and maintenance costs, which are the major drivers in DOC. Tables LVII and LVIII also show the price of the current technology engine with learning in a footnote to the table.

TABLE LVII. - STAT TURBOPROP ENGINE COMPARISONS (SI UNITS)

	'CTE'	DTE	ATE
	Rated power = 1749 kW		
*sfc--μg/W·s	86.19	76.46	71.69
Percent change	0	-11.3	-16.8
Weight--kg	364.7	358.3	318.9
Percent change	0	-1.7	-12.6
OEM price--$	352,800**	371,200	286,000
Percent change	0	+5.2	-18.9
Maintenance cost--$/EFH	40.73	37.38	17.87
Percent change	0	-8.2	-56.1
Max envelope length--m	2.292	2.217	1.793
Percent change	0	-3.3	-21.8
Max envelope height--m	0.705	0.675	0.668
Percent change	0	-4.2	-5.3
	Rated power = 3544 kW		
*sfc--μg/W·s	79.74	70.78	64.68
Percent change	0	-11.2	-19.0
Weight--kg	533.9	524.8	398.7
Percent change	0	-1.7	-25.3
OEM price--$	560,100***	599,200	468,500
Percent change	0	+7.0	-16.4
Maintenance cost--$/EFH	64.91	59.57	24.64
Percent change	0	-8.2	-62.0
Max envelope length--m	2.658	2.572	2.215
Percent change	0	-3.3	-16.7
Max envelope height--m	0.847	0.811	0.735
Percent change	0	-4.2	-13.3

*Gearbox loss included
**CTE OEM price = $285,500
***CTE OEM price = $430,800

TABLE LVIII. - STAT TURBOPROP ENGINE COMPARISONS (CUSTOMARY UNITS)

	'CTE'	DTE	ATE
	Rated power = 2345 shp		
*sfc--lbm/hp-hr	0.5101	0.4525	0.4243
Percent change	0	-11.3	-16.8
Weight--lbm	804	790	703
Percent change	0	-1.7	-12.6
OEM price--$	352,800**	371,200	286,000
Percent change	0	+5.2	-18.9
Maintenance cost--$/EFH	40.73	37.38	17.87
Percent change	0	-8.2	-56.1
Max envelope length--in.	90.23	87.29	70.6
Percent change	0	-3.3	-21.8
Max envelope height--in.	27.74	26.58	26.28
Percent change	0	-4.2	-5.3
	Rated power = 4752 shp		
*sfc--lbm/hp-hr	0.4719	0.4189	0.3823
Percent change	0	-11.2	-19.0
Weight--lbm	1177	1157	879
Percent change	0	-1.7	-25.3
OEM price--$	560,100***	599,200	468,500
Percent change	0	+7.0	-16.4
Maintenance cost--$/EFH	64.91	59.57	24.64
Percent change	0	-8.2	-62.0
Max envelope length--in.	104.66	101.25	87.2
Percent change	0	-3.3	-16.7
Max envelope height--in.	33.34	31.94	28.92
Percent change	0	-4.2	-13.3

*Gearbox loss included
**CTE OEM price = $285,500
***CTE OEM price = $430,800

Mission Results

The engine technology comparisons will present results from the mission and aircraft cost analysis of the current, derivative, and advanced technology engines having characteristics as developed and presented in previous sections of this report. It is noted that in this study, aircraft characteristics and technology level are fixed; i.e., each engine technology is evaluated without change in aircraft technology, even though it is probable that advancements in areas such as aerodynamics and structures could be incorporated with advanced engines to provide additional savings. This is also true with respect to the propeller characteristics; i.e., each engine technology incorporates current conventional propeller efficiencies, weight, acquisition, and maintenance cost values. Again, it is likely that advanced technology propellers would provide additional cost savings.

Engine performance data for the derivative and advanced engines were simulated by using sfc characteristics of the DDA current technology engine (Model PD370-37). The simulation consists of adjusting the sfc level of the CTE by the ratio of the design point sfc values for the respective derivative and advanced engine to the design point sfc value of the CTE engine.

High Speed Aircraft

The following section presents a general overview of the results obtained from mission evaluations of the CTE, DTE, and ATE in the high speed aircraft. This overview will be followed by a more detailed presentation of the technology comparisons obtained from the 185.2 km (100 nm) alternate stage length mission results.

General mission results, i.e., aircraft design TOGW, design fuel load, engine take-off rating, and total aircraft cost (TAC) are summarized in Tables LIX and LX along with fuel consumption, DOC, and 5-year total cost of ownership (TCO) results for the 1111.2 km (600 nm) design stage length and the 185.2 km (100 nm) alternate stage length missions. Tables LXI and LXII lists the DOC and TCO results for each of the alternate stage lengths, i.e., 97.6-, 185.2-,. 277.8-, 370.4-, and 740.8 km (50-, 100-, 150-, 200-, and 400 nm) missions.

TABLE LIX. - MISSION RESULTS--HIGH SPEED AIRCRAFT (SI UNITS)

	CTE	'CTE'	DTE	ATE
Design aircraft				
TOGW--kg	18,299	18,299	17,895	17,193
Fuel--kg	1940	1940	1700	1500
Fuel fraction	0.106	0.106	0.095	0.087
Engine TO rating at SLSS--kW	3532	3532	3467	3341
Aircraft power loading--kW/kg	0.386	0.386	0.388	0.388
Airframe acquisition--million $	4.226	4.226	4.198	4.148
Propulsion system acquisition--million $	1.609	2.042	2.144	1.666
Total aircraft cost--million $	5.835	6.268	6.342	5.814
1111.2 km stage length (design)				
Design cruise speed--M_N	0.7	0.7	0.7	0.7
Cruise altitude--m	10,668	10,668	10,668	20,668
Block fuel--kg	1190	1190	1041	920
Fuel consumption--L x 10^{-3} aircraft/year	2496	2496	2184	1931
DOC at ① --¢/skm	2.355	2.406	2.286	1.932
DOC at ② --¢/skm	2.718	2.769	2.603	2.213
5-y TCO at ① --million $	14.249	14.745	14.245	12.320
5-y TCO at ② --million $	15.897	16.393	15.687	13.595

TABLE LIX. (CONT)

	CTE	'CTE'	DTE	ATE
185.2 km stage length (altitude)				
Cruise altitude--m	5791	5791	5791	5791
Block fuel--kg	380	380	336	293
Fuel consumption--L x 10^{-3}/aircraft/year	3155	3155	2807	2435
DOC at 1--¢/skm	4.794	4.871	4.643	4.066
DOC at 2--¢/skm	5.490	5.567	5.258	4.603
5-y TCO at 1--million $	17.902	18.398	17.857	15.717
5-y TCO at 2--million $	19.986	20.481	19.711	17.324

1 fuel cost = $0.264/L
2 fuel cost = $0.396/L

TABLE LX. - MISSION RESULTS--HIGH SPEED AIRCRAFT (CUSTOMARY UNITS)

	CTE	'CTE'	DTE	ATE
Design aircraft				
TOGW--lbm	40,343	40,343	39,453	37,905
Fuel--lbm	4276	4276	3747	3308
Fuel fraction	0.106	0.106	0.095	0.087
Engine TO rating at SLSS--shp	4736	4736	4649	4480
Aircraft power loading--shp/lbm	0.235	0.235	0.236	0.236
Airframe acquisition--million $	4.226	4.226	4.198	4.148
Propulsion system acquisition--million $	1.609	2.042	2.144	1.666
Total aircraft cost--million $	5.835	6.268	6.342	5.814
600-NM stage length (design)				
Design cruise speed--M_N	0.7	0.7	0.7	0.7
Cruise altitude--ft	35,000	35,000	35,000	35,000
Block fuel--lbm	2623	2623	2295	2029
Fuel consumption--gal x 10^{-3}/aircraft/year	659.3	659.3	576.9	510.1
DOC at 1--¢/snm	4.361	4.455	4.233	3.578
DOC at 2--¢/snm	5.033	5.128	4.821	4.098
5-y TCO at 1--million $	14.249	14.745	14.245	12.320
5-y TCO at 2--million $	15.897	16.393	15.687	13.595
100-NM stage length (altitude)				
Cruise altitude--ft	19,000	19,000	19,000	19,000
Block fuel--lbm	838	838	741	647
Fuel consumption--gal x 10^{-3}/aircraft/year	833.4	833.4	741.4	643.1
DOC at 1--¢/snm	8.878	9.021	8.598	7.530
DOC at 2--¢/snm	10.168	10.311	9.738	8.524
5-y TCO at 1--million $	17.902	18.398	17.857	15.717
5-y TCO at 2--million $	19.986	20.481	19.711	17.324

1 fuel cost = $1.00/gal
2 fuel cost = $1.50/gal

TABLE LXI. - ALTERNATE STAGE LENGTH MISSION RESULTS--HIGH SPEED AIRCRAFT
(SI UNITS)

	CTE	'CTE'	DTE	ATE
92.6 km stage length				
Cruise altitude--m	3048	3048	3048	3048
Block fuel--kg	258.1	258.1	225.4	199.1
DOC--¢/skm--①	7.326	7.431	7.118	6.321
DOC--¢/skm--②	8.271	8.409	7.943	7.050
5-y TCO--million $--①	19.595	20.091	19.452	17.387
5-y TCO--million $--②	21.663	22.158	21.259	18.984
185.2 km stage length				
Cruise altitude--m	5791	5791	5486	5791
Block fuel--kg	380.1	380.1	336.1	293.5
DOC--¢/skm--①	4.794	4.871	4.643	4.066
DOC--¢/skm--②	5.490	5.567	5.258	4.603
5-y TCO--million $--①	17.902	18.398	17.857	15.717
5-y TCO--million $--②	19.986	20.481	19.711	17.324
277.8 km stage length				
Cruise altitude--m	7315	7315	7010	7315
Block fuel--kg	477.2	477.2	421.4	367.9
DOC--¢/skm--①	3.873	3.940	3.750	3.254
DOC--¢/skm--②	4.455	4.523	4.264	3.704
5-y TCO--million $--①	16.860	17.356	16.828	14.729
5-y TCO--million $--②	18.860	19.356	18.606	16.272
370.4 km stage length				
Cruise altitude--m	8230	8230	7925	8230
Block fuel--kg	565.6	565.6	499.4	436.4
DOC--¢/skm--①	3.390	3.452	3.282	2.832
DOC--¢/skm--②	3.908	3.970	3.739	3.231
5-y TCO--million $--①	16.188	16.683	16.166	14.098
5-y TCO--million $--②	18.177	18.612	17.878	15.587
740.8 km stage length				
Cruise altitude--m	10363	10058	10058	10058
Block fuel--kg	877.2	884.5	772.9	683.1
DOC--¢/skm--①	2.624	2.678	2.545	2.165
DOC--¢/skm--②	3.026	3.083	2.898	2.478
5-y TCO--million $--①	14.773	15.316	14.788	12.839
5-y TCO--million $--②	16.489	17.052	16.306	14.180

① fuel cost = $0.264/L
② fuel cost = $0.396/L

TABLE LXII. - ALTERNATE STAGE LENGTH MISSION RESULTS--HIGH SPEED AIRCRAFT (CUSTOMARY UNITS)

	CTE	'CTE'	DTE	ATE
50-NM stage length				
Cruise altitude--ft	10,000	10,000	10,000	10,000
Block fuel--lbm	569	569	497	439
DOC--¢/snm--①	13.567	13.763	13.182	11.706
DOC--¢/snm--②	15.317	15.513	14.711	13.056
5-y TCO--million $--①	19.595	20.091	19.452	17.387
5-y TCO--million $--②	21.663	22.158	21.259	18.984
100-NM stage length				
Cruise altitide--ft	19,000	19,000	18,000	19,000
Block fuel--lbm	838	838	741	674
DOC--¢/snm--①	8.878	9.021	8.598	7.530
DOC--¢/snm--②	10.168	10.311	9.738	8.524
5-y TCO--million $--①	17.902	18.398	17.857	15.717
5-y TCO--million $--②	19.986	20.481	19.711	17.324
150-NM stage length				
Cruise altitude--ft	24,000	24,000	23,000	24,000
Block fuel--lbm	1052	1052	929	811
DOC--¢/snm--①	7.172	7.297	6.945	6.027
DOC--¢/snm--②	8.251	8.376	7.897	6.859
5-y TCO--million $--①	16.860	17.356	16.828	14.729
5-y TCO--million $--②	18.860	19.356	18.606	16.272
200-NM stage length				
Cruise altitude--ft	27,000	27,000	26,000	27,000
Block fuel--lbm	1247	1247	1101	962
DOC--¢/snm--①	6.278	6.394	6.079	5.244
DOC--¢/snm--②	7.238	7.353	6.925	5.984
5-y TCO--million $--①	16.188	16.683	16.166	14.098
5-y TCO--million $--②	18.117	18.612	17.878	15.587
400-NM stage length				
Cruise altitude--ft	34,000	33,000	33,000	33,000
Block fuel--lbm	1934	1950	1704	1506
DOC--¢/snm--①	4.860	4.960	4.713	4.010
DOC--¢/snm--②	5.604	5.710	5.368	4.589
5-y TCO--million $--①	14.773	15.316	14.788	12.839
5-y TCO--million $--②	16.489	17.052	16.306	14.180

① fuel cost = $1.00/gal
② fuel cost = $1.50/gal

Note that Tables LIX through LXII list DOC and TCO results for fuel costs of $0.264 and $0.396/L ($1.00 and $1.50/gal).

The DOC results listed in Tables LIX through LXII are illustrated in Figures 47 and 48 as plots of DOC versus block distance (mission stage length) for the $0.264 and $0.396/L ($1.00 and $1.50/gal) fuel costs, respectively. Figure 49 shows the percent reductions in DOC obtained with the DTE and ATE, relative to the CTE, plotted against block distance. The DOC savings for the DTE are indicated in Figure 49 to be relatively insensitive to block distance with an approximate 4% reduction at the $0.396/L ($1.50/gal) fuel cost. The DOC results for the ATE indicate 15.5% reduction at the higher fuel cost increasing to 18.5% at greater block distances.

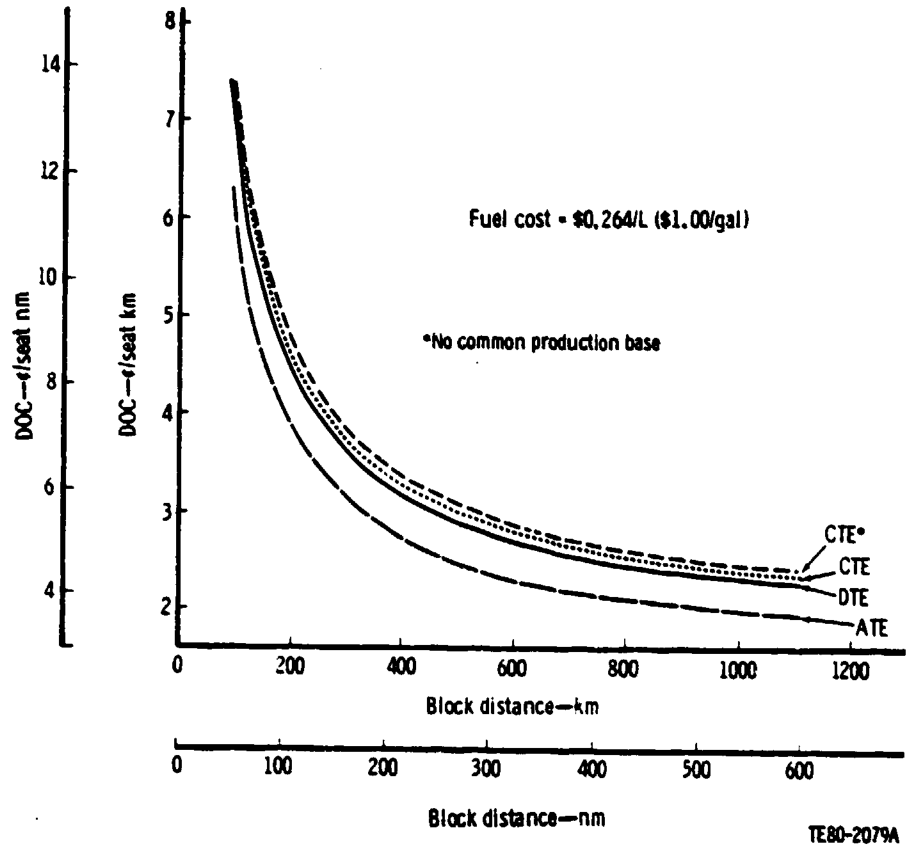


Figure 47. - DOC versus block distance--high speed aircraft, fuel cost $0.264/L ($1.00/gal).

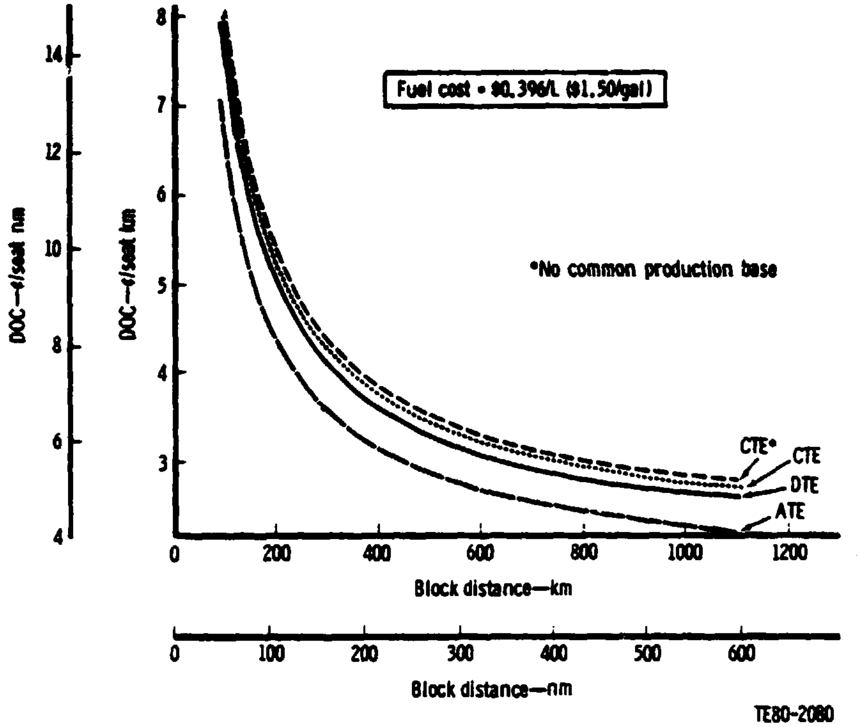


Figure 48. - DOC versus block distance--high speed aircraft, fuel cost $0.396/L ($1.50/gal).

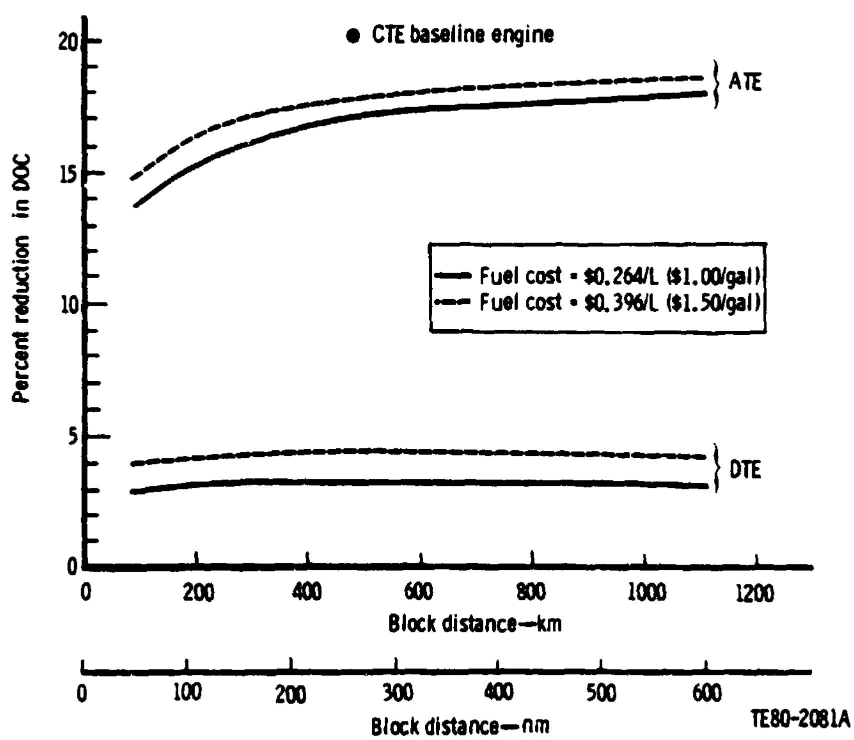


Figure 49. - DOC reductions--high speed aircraft.

The 5-year TCO results listed in Tables LIX through LXII are shown in Figures 50 and 51 as plots of TCO versus block distance. Figure 52 shows the percent reductions in TCO obtained with the DTE and ATE, relative to the CTE, plotted against block distance. The percent reductions in TCO for both the DTE and ATE are indicated to be relatively insensitive to block distance with the DTE having an approximate 1% reduction and the ATE a 14% reduction in TCO at the \$0.396/L (\$1.50/gal) fuel cost. Figure 53 shows the 5-year TCO improvement obtained with the DTE and ATE in terms of dollar savings at each block distance. The DTE indicates an approximate \$250,000 savings at each block distance with the \$0.396/L (\$1.50/gal) fuel cost. The ATE indicates an approximate \$2.3 to \$2.7 million savings dependent upon block distance, with the \$0.396/L (\$1.50/gal) fuel.

The fuel burned or fuel consumption improvements obtained with the DTE and ATE are illustrated in Figure 54. This figure shows percent reduction in fuel consumption, relative to the CTE, plotted against block distance. The reductions for both the DTE and ATE are indicated to be essentially constant over the range of block distances, with the DTE obtaining an approximate 11 to 12% reduction, and the ATE an approximate 22 to 23% reduction in fuel burned per mission. The fuel savings that could be achieved over a 10-year period of operation with a fleet of 100 aircraft is shown in Figure 55. This figure plots DTE and ATE fuel savings, relative to the CTE, versus block distance. Figure 55 indicates relatively higher savings to the shorter block distance with the DTE savings ranging approximately from 341 to 303 million liters (90 to 80 million gal). The fuel savings obtained with the ATE range approximately from 719 to 568 million liters (190 to 150 million gal) of fuel.

The variation in block speed across the range of block distances examined in this study is shown in Figure 56. This figure shows a change in block speed from 315 km/h (170 kt) for the 92.6 km (50 nm) distance to 648 km/h (350 kt) for the 1111.2 km (600 nm) design stage length with the block speed for the 185.2 km (100 nm) distance indicated to be approximately 426 km/h (230 kt).

A summary of the percentage changes in critical aircraft and cost parameters resulting from engine technology improvements is shown in Table LXIII. These percentage changes are relative to the CTE powered aircraft values. The cost parameters were calculated for the 185.2 km (100 nm) stage length mission (185.2 km (100 nm) block distance) only. (Note that 'CTE' is the same engine as CTE except for price, which is higher. 'CTE' reflects an all new engine with no inherited learning.)

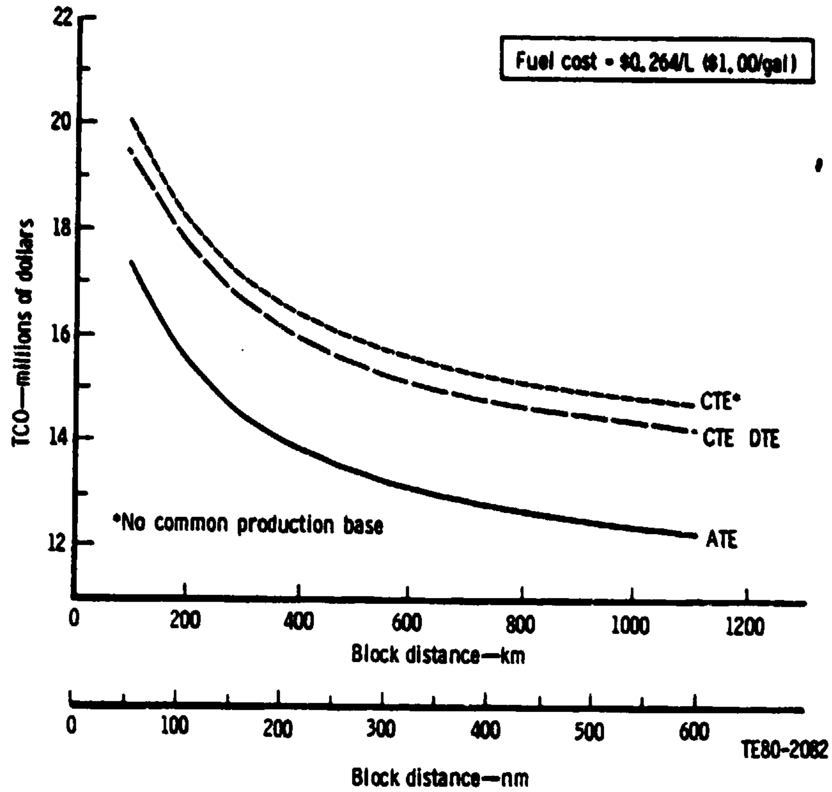


Figure 50. - 5-year TCO versus block distance--high speed aircraft, fuel cost $0.264/L ($1.00/gal).

TABLE LXIII. - TECHNOLOGY COMPARISON--HIGH SPEED AIRCRAFT

185.2 km (100 nm) block distance

		Percent change from CTE		
	CTE	'CTE'	DTE	ATE
TOGW	Base	0	-2.2	-6.0
TAC	Base	+7.4	+8.7	-0.4
Fuel consumption	Base	0	-11.0	-22.8
DOC at ①	Base	+1.6	-3.2	-15.2
DOC at ②	Base	+1.4	-4.2	-16.2
5-y TCO at ①	Base	+2.8	-0.2	-12.2
5-y TCO at ②	Base	+2.5	-1.4	-13.3

① fuel cost = $0.264/L ($1.00/gal)
② fuel cost = $0.396/L ($1.50/gal)
Utilization = 2800 h/y

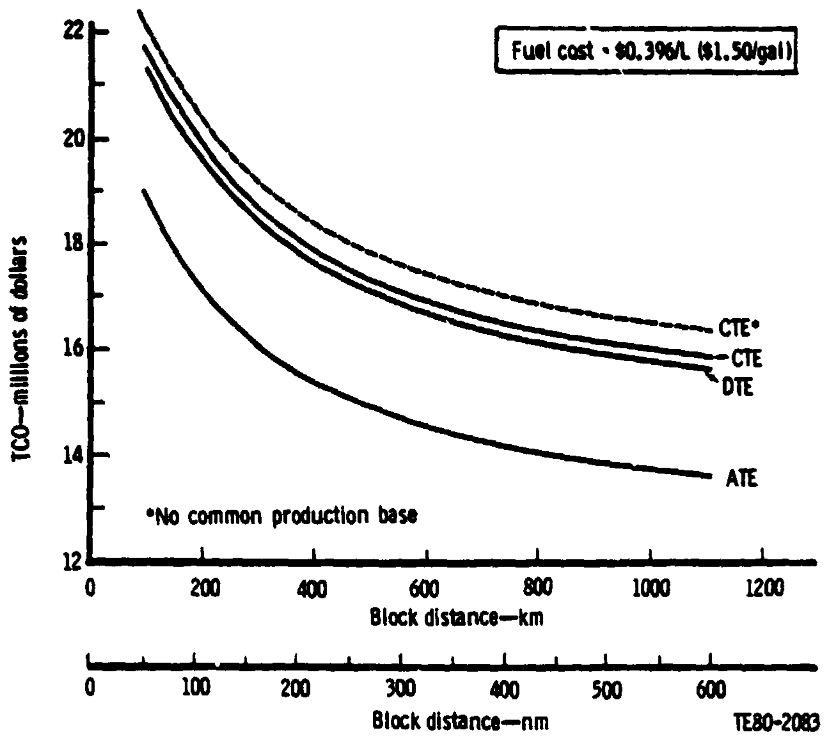


Figure 51. - 5-year TCO versus block distance--high speed aircraft, fuel cost $0.396/L ($1.50/gal).

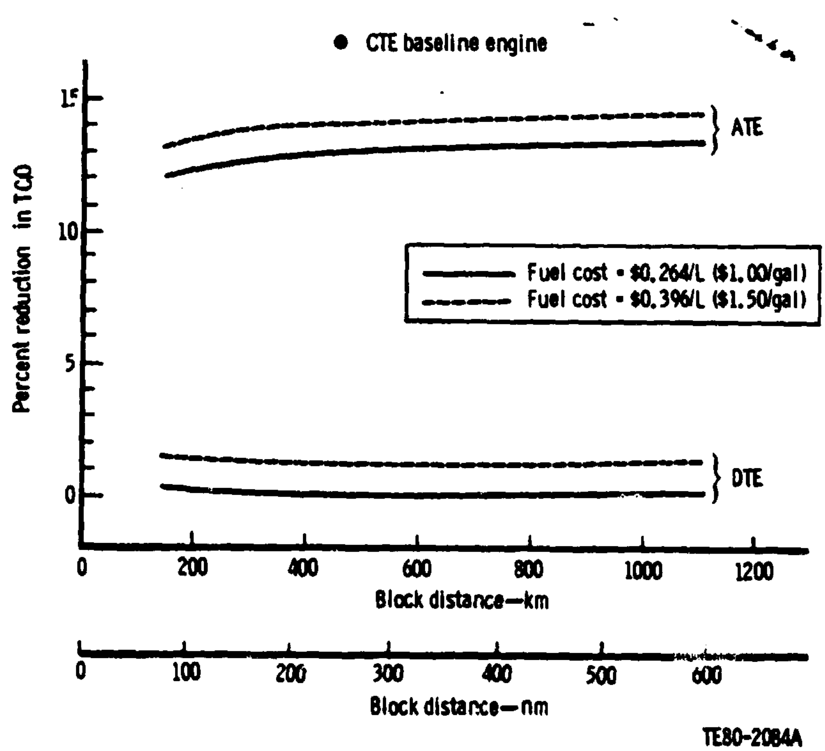


Figure 52. - 5-year TCO reductions--high speed aircraft.

The combination of the sfc, weight, and envelope dimensions of the ATE is shown to have reduced aircraft TOGW by 6%. However, the total aircraft cost is shown to be essentially the same as the CTE-powered aircraft. This is a result of the fact that the lower airframe less power section cost was offset by the higher acquisition cost of the ATE compared to the CTE. It is noted that the significantly higher cost of the DTE with respect to the ATE con tributed significantly to a 9% increase in total aircraft cost over the CTE.

The fuel consumption percentage changes basically reflect the sfc improvements associated with the DTE and ATE plus the reduced rated engine power associated with the CTE and ATE aircraft, which in turn resulted from the gross weight reduction. The 12% difference between the DOC and TCO reductions shown for the DTE and those shown for the ATE is a result of the combination of the ATE's larger sfc, engine acquisition, and maintenance cost improvement relative to the DTE. It is noted that increasing the fuel cost from $0.264 to $0.396/l ($1.00 to $1.50/gal) produces a 1% larger reduction in DOC and TCO for both the derivative and advanced engines compared to the current engine.

A comparison of the 10-year fuel requirements of an assumed fleet of 100 aircraft flying the 185.2 km (100 nm) stage length exclusively is shown in the bar graph presented in Figure 57. This figure indicates a savings of 348 million liters (92 million gal) for the DTE, and 719 million liters (190 million gal) of fuel for the ATE.

A comparison of the 10-year DOC for the fleet and mission prescribed in the preceding fuel comparison is shown in Figure 58 for both $0.264 and $0.396/liter ($1.00 and $1.50/gal) fuel costs. This bar graph indicated a savings in the order of $116 million for the DTE, and $529 million for the ATE at the $0.396/liter ($1.50/gal) fuel cost.

A breakdown of the cost element in DOC for each of the technology engines is shown in Figure 59. Figure 59 indicates that the DOC reduction for the DTE was essentially a result of the sfc improvement. Note that the sfc improvement was offset to a small extent by the higher depreciation and insurance costs associated with the higher engine acquisition cost. The DOC reduction noted for the ATE was essentially a result of a combination of significant sfc reduction, engine acquisition, and maintenance cost improvements. This figure also illustrates that significant improvement in efficiency is required to maintain current DOC levels in the face of rising fuel costs.

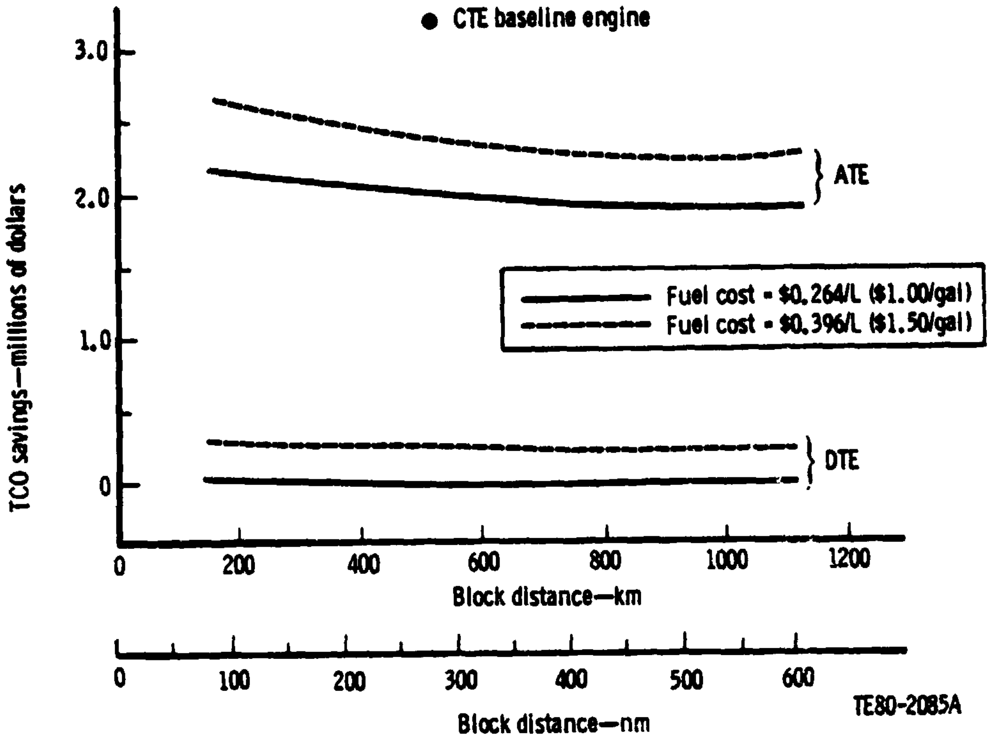


Figure 53. - 5-year TCO savings--high speed aircraft.

Low Speed Aircraft

The following section presents a general overview of the results obtained from mission evaluations of the CTE, DTE, and ATE in the DDA low speed aircraft. This overview will be followed by a more detailed presentation of the technology comparisons obtained from the 185.2 km (100 nm) alternate stage length mission results.

General mission results, i.e., aircraft design TOGW, design fuel load, engine take-off rating, and TAC, are summarized in Tables LXIV and LXV along with fuel consumption, DOC, and 5-year TCO results for the 1111.2 km (600 nm) design stage length and the 185.2 km (100 nm) alternate stage length missions. Tables LXVI and LXVII list the DOC and TCO results for each of the alternate stage lengths.

Note that Table LXIV through LXVII list DOC and TCO results for fuel costs of $0.264 and $0.396/L ($1.00 and $1.50/gal).

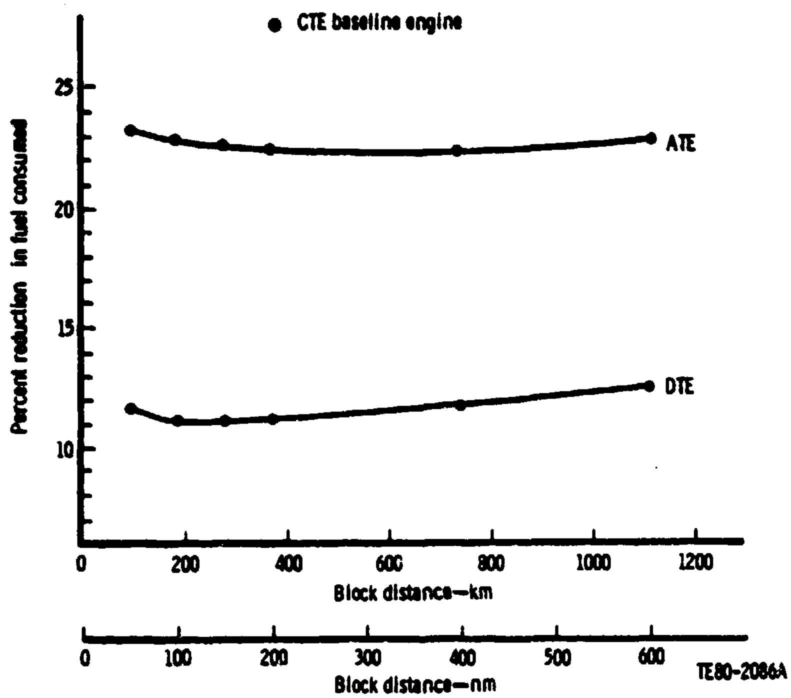


Figure 54. - Percent reduction in fuel consumed--high speed aircraft.

TABLE LXIV. - MISSION RESULTS--LOW SPEED AIRCRAFT (SI UNITS)

	CTE	'CTE'	DTE	ATE
Design aircraft				
TOGW--kg	18,928	18,928	18,468	18,083
Fuel--kg	2080	2080	1822	1684
Fuel fraction	0.110	0.110	0.099	0.093
Engine TO rating at SLSS--kw	2139	2139	2095	2057
Aircraft power loading--kW/kg	0.083	0.083	0.084	0.084
Airframe acquisition--million $	3.986	3.986	3.954	3.926
Propulsion system acquisition--million $	1.116	1.421	1.468	1.138
Total aircarft cost--million $	5.102	5.407	5.422	5.064
1111.2-km stage length (design)				
Design cruise speed--M_N	0.47	0.47	0.47	0.47
Cruise altitude--m	6096	6096	6096	6096
Block fuel--kg	1466	1466	1279	1180
Fuel consumption--L x 10^{-3}/aircraft/year	2298	2298	2005	1851
DOC at 1 --¢/skm	2.949	2.997	2.846	2.544
DOC at 2 --¢/skm	3.396	3.444	3.237	2.904
5-y TCO at 1 --million $	13.118	13.467	12.966	11.722
5-y TCO at 2 --million $	14.636	14.985	14.290	12.944

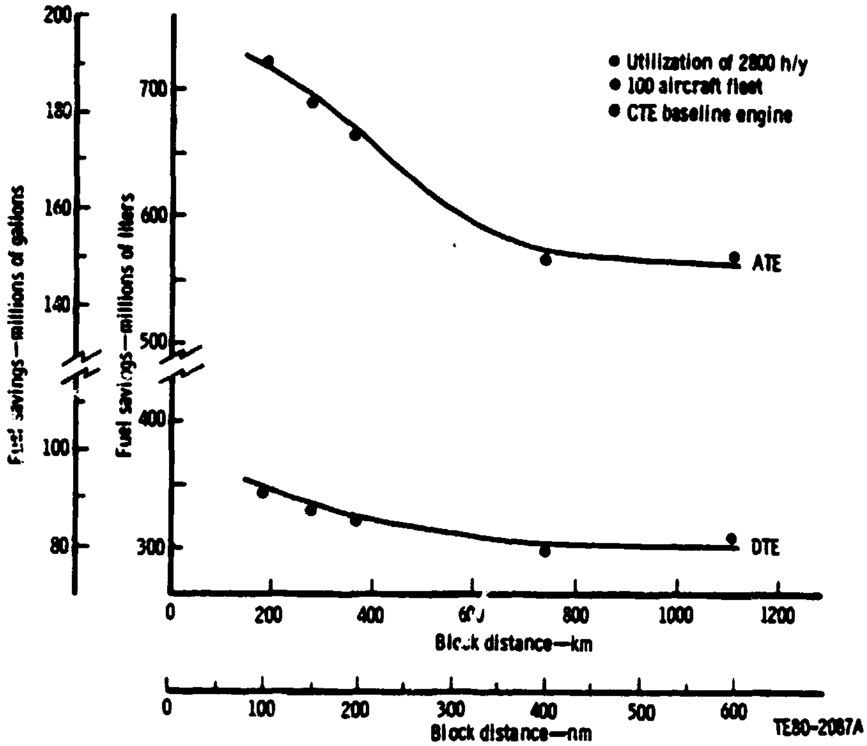


Figure 55. - 10-year fuel savings--high speed aircraft.

TABLE LXIV. (CONT)

	CTE	'CTE'	DTE	ATE
185.2-km stage length (altitude)				
Cruise altitude--m	3048	3048	3048	3048
Block fuel--kg	369	369	321	296
Fuel consumption--L x 10^{-3}/aircraft/year	2523	2523	2199	2028
DOC at ① --¢/skm	4.030	4.096	3.884	3.522
DOC at ② --¢/skm	4.705	4.771	4.472	4.064
5-y TOC at ① --million $	13.063	13.412	12.900	11.789
5-y TCO at ② --million $	14.730	15.078	14.352	13.128

① fuel cost = $0.264/L
② fuel cost = $0.396/L

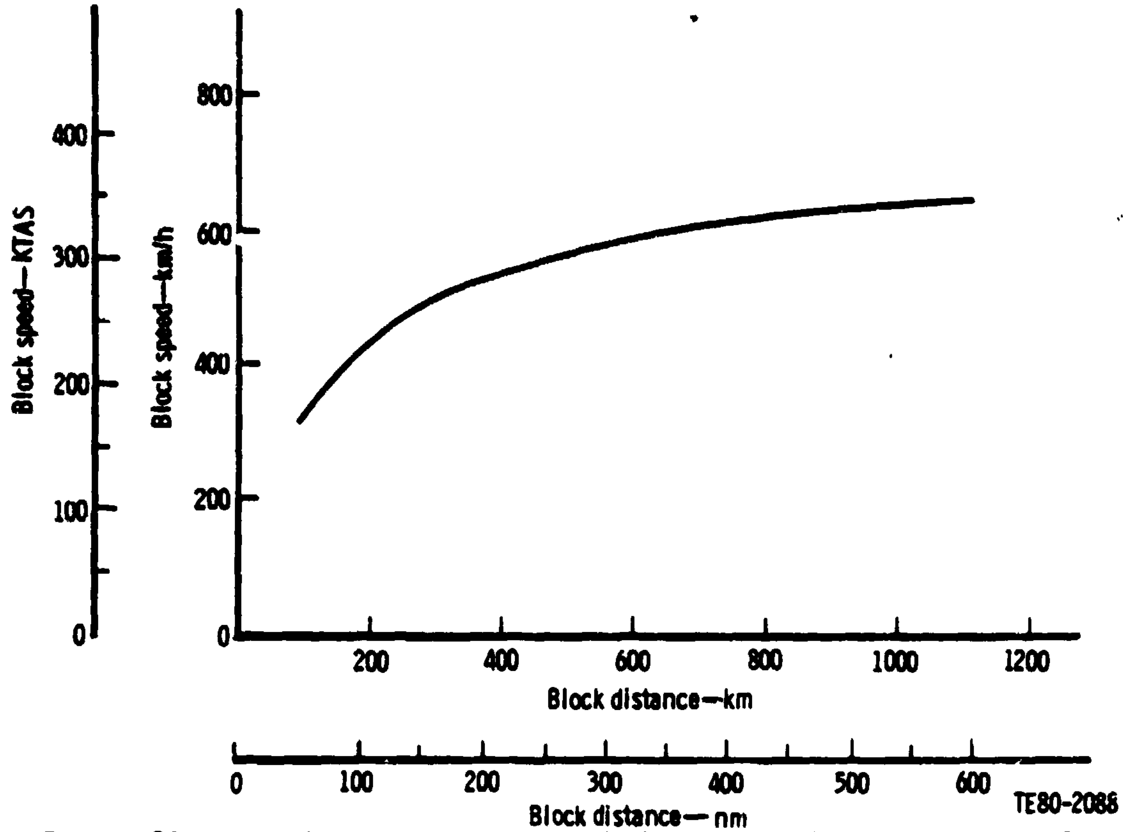


Figure 56. - Block speed versus block distance--high speed aircraft.

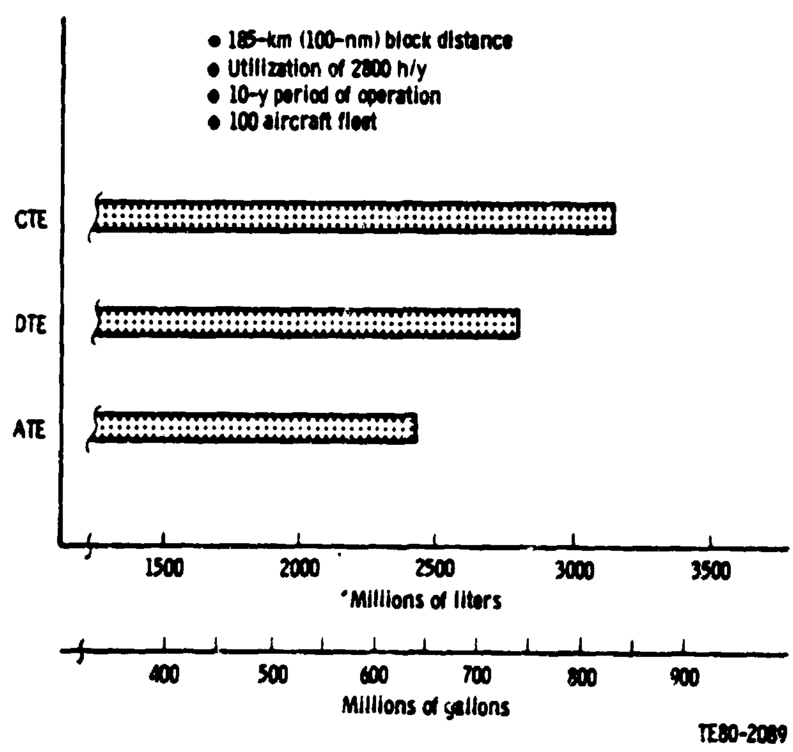


Figure 57. - Fuel consumption comparison--high speed aircraft.

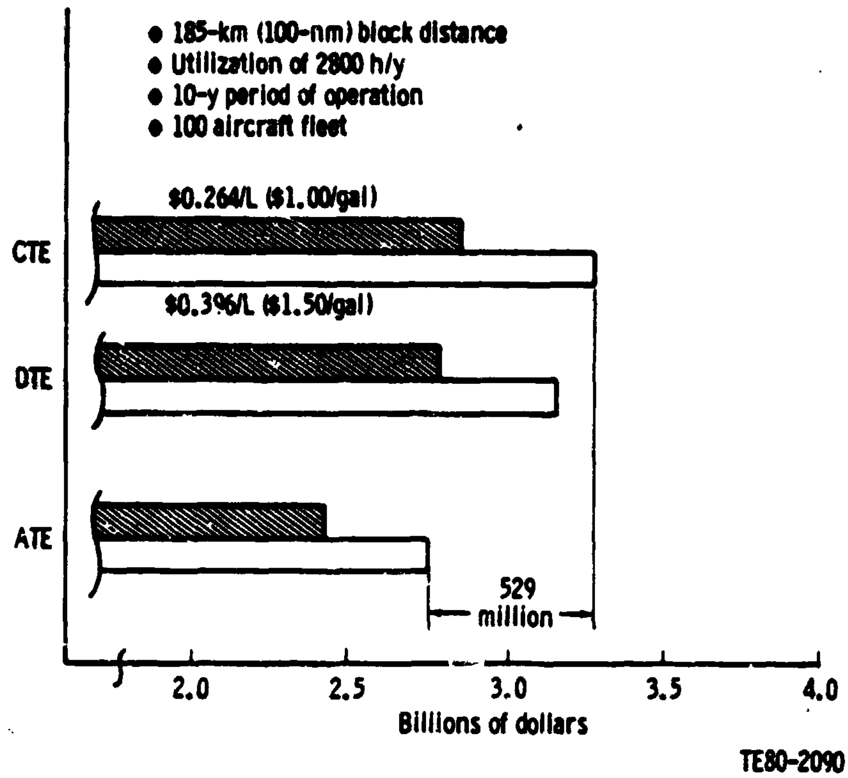


Figure 58. - DOC comparison--high speed aircraft.

TABLE LXV. - MISSION RESULTS--LOW SPEED AIRCRAFT (CUSTOMARY UNITS)

	CTE	'CTE'	DTE	ATE
Design aircraft				
TOGW--lbm	41,730	41,730	40,715	39,866
Fuel--lbm	4586	4586	4016	3713
Fuel fraction	0.110	0.110	0.099	0.093
Engine TO rating at SLSS--shp	2868	2868	2810	2758
Aircraft power loading--shp/lbm	0.137	0.137	0.138	0.138
Airframe acquisition--million $	3.986	3.986	3.954	3.926
Propulsion system acquisition--million $	1.116	1.421	1.468	1.138
Total aircraft cost--million $	5.102	5.407	5.422	5.064
600-NM stage length (design)				
Design cruise speed--M_N	0.47	0.47	0.47	0.47
Cruise altitude--ft	20,000	20,000	20,000	20,000
Block fuel-lbm	3231	3231	2819	2602
Fuel consumption--gal x 10^{-3}/aircraft/year	607.0	607.0	529.7	488.9
DOC at ① --¢/snm	5.461	5.550	5.271	4.711
DOC at ② --¢/snm	6.289	6.378	5.994	5.378
5-y TCO at ① --million $	13.118	13.467	12.966	11.722
5-y TCO at ② --million $	14.636	14.985	14.290	12.944

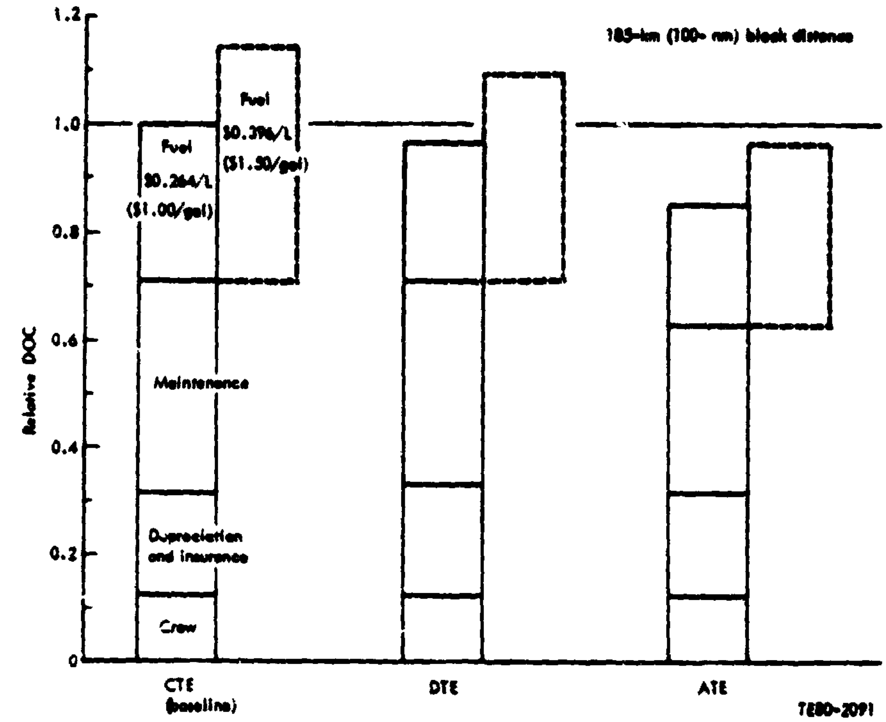


Figure 59. - DOC breakdown--high speed aircraft.

TABLE LXV. (CONT)

	CTE	'CTE'	DTE	ATE
100-NM stage length (altitude)				
Cruise altitude--ft	10,000	10,000	10,000	10,000
Block fuel--lbm	813	813	708	653
Fuel consumption--gal x 10^{-3}/aircraft/year	666.6	666.6	581.0	535.8
DOC at ① --¢/snm	7.463	7.585	7.193	6.523
DOC at ② --¢/snm	8.713	8.835	8.282	7.527
5-y TCO at ① --million $	13.063	13.412	12.900	11.789
5-y TCO at ② --million $	14.730	15.078	14.352	13.128

① fuel cost = $1.00/gal
② fuel cost = $1.50/gal

TABLE LXVI. - ALTERNATE STAGE LENGTH MISSION RESULTS--LOW SPEED AIRCRAFT
(SI UNITS)

	CTE	'CTE'	DTE	ATE
92.6-km stage length				
Cruise altitude--m	3048	3048	3048	3048
Block fuel--kg	219	219	191	176
DOC--¢/skm--①	5.261	5.352	5.092	4.614
DOC--¢/skm--②	6.063	6.153	5.790	5.258
5-y TCO--million $--①	12.629	12.978	12.520	11.438
5-y TCO--million $--②	14.078	14.427	13.782	12.602
185.2-km stage length				
Cruise altitude--m	3048	3048	3048	3048
Block fuel--kg	369	369	321	296
DOC--¢/skm--①	4.030	4.096	3.884	3.522
DOC--¢/skm--②	4.705	4.771	4.472	4.064
5-y TCO--million $--①	13.063	13.412	12.900	11.789
5-y TCO--million $--②	14.730	15.078	14.352	13.128
277.8-km stage length				
Cruise altitude--m	3962	3962	3658	3962
Block fuel--kg	495	495	437	398
DOC--¢/skm--①	3.606	3.666	3.476	3.152
DOC--¢/skm--②	4.211	4.270	4.010	3.638
5-y TCO--million $--①	13.066	13.415	12.961	11.791
5-y TCO--million $--②	14.734	15.082	14.444	13.132
370.4-km stage length				
Cruise altitude--m	4572	4572	4572	4572
Block fuel-kg	611	611	533	491
DOC--¢/skm--①	3.494	3.549	3.363	3.007
DOC--¢/skm--②	4.053	4.109	3.850	3.457
5-y TCO--million $--①	13.369	13.718	13.181	11.918
5-y TCO--million $--②	15.011	15.360	14.613	13.239
740.8-km stage length				
Cruise altitude--m	6096	6096	6096	6096
Block fuel--kg	1029	1029	897	828
DOC--¢/skm--①	3.093	3.143	2.984	2.667
DOC--¢/skm--②	3.564	3.614	3.395	3.046
5-y TCO--million $--①	13.137	13.486	12.981	11.734
5-y TCO--million $--②	14.664	15.013	14.312	12.963

① fuel cost = $0.264/L
② fuel cost = $0.396/L

TABLE LXVII. - ALTERNATE STAGE LENGTH MISSION RESULTS--LOW SPEED AIRCRAFT
(CUSTOMARY UNITS)

	CTE	'CTE'	DTE	ATE
50-NM stage length				
Cruise altitide--ft	10,000	10,000	10,000	10,000
Block fuel-lbm	482	482	420	387
DOC--¢/snm--①	9.744	9.911	9.430	8.545
DOC--¢/snm--②	11.228	11.395	10.723	9.737
5-y TCO--million $--①	12.629	12.978	12.520	11.438
5-y TCO--million $--②	14.078	14.427	13.782	12.602
100-NM stage length				
Cruise altitude--ft	10,000	10,000	10,000	10,000
Block fuel-lbm	813	813	708	653
DOC--¢/snm--①	7.463	7.585	7.193	6.523
DOC--¢/snm--②	8.713	8.835	8.282	7.527
5-y TCO--million $--①	13.063	13.412	12.900	11.789
5-y TCO--million $--②	14.730	15.078	14.352	13.128
150-NM stage length				
Cruise altitude--ft	13,000	13,000	12,000	13,000
Block fuel--lbm	1092	1092	964	877
DOC--¢/snm--①	6.679	6.789	6.437	5.838
DOC--¢/snm--②	7.799	7.908	7.426	6.738
5-y TCO--million $--①	13.066	13.415	12.961	11.791
5-y TCO--million $--②	14.734	15.082	14.444	13.132
200-NM stage length				
Cruise altitude--ft	15,000	15,000	15,000	15,000
Block fuel--lbm	1347	1347	1174	1083
DOC--¢/snm--①	6.470	6.573	6.228	5.569
DOC--¢/snm--②	7.506	7.609	7.131	6.402
5-y TCO--million $--①	13.369	13.718	13.181	11,918
5-y TCO--million $--②	15.011	15.360	14.613	13.239
400-NM stage length				
Cruise altitude--ft	20,000	20,000	20,000	20,000
Block fuel--lbm	2269	2269	1978	1825
DOC--¢/snm--①	5.728	5.821	5.526	4.939
DOC--¢/snm--②	6.600	6.693	6.287	5.641
5-y TCO--million $--①	13.137	13.486	12.981	11.734
5-y TCO--million $--②	14.664	15.013	14.312	12.963

① fuel cost = $1.00/gal
② fuel cost = $1.50/gal

The DOC results listed in Tables LXIV through LXVII are illustrated in Figures 60 and 61 as plots of DOC versus block distance (mission stage length) for the \$0.264 and \$0.396/L (\$1.00 and \$1.50/gal) fuel costs, respectively. Figure 62 shows the percent reductions in DOC obtained with the DTE and ATE, relative to the CTE, plotted against block distance. The DOC savings for the DTE are indicated in Figure 62 to be relatively insensitive to block distance with an approximate 5% reduction at the \$0.396/L (\$1.50/gal) fuel cost. The DOC results for the ATE indicate slightly lower percentage reductions for the shorter block the ATE indicate slightly lower percentage reductions for the shorter block distances. A range of 13 to 15% reduction at the higher fuel cost is shown.

The 5-year TCO results listed in Tables LXIV through LXVII are shown in Figures 63 and 64 as plots of TCO versus block distance. Figure 65 shows the percent reductions in TCO obtained with the DTE and ATE, relative to the CTE, plotted against block distance. The percent reductions in TCO for both the DTE and ATE are indicated to be relatively insensitive to block distance with the DTE having an approximate 2.5% reduction and the ATE a 11% reduction in TCO at the \$0.396/L (\$1.50/gal) fuel cost. Figure 66 shows the 5-year TCO improvement obtained with the DTE and ATE in terms of dollar savings at each block distance. The DTE indicates an approximate \$200,000 savings at each block distance with the \$0.396/L (\$1.50/gal) fuel cost. The ATE indicates an approximate \$1.5 to 1.7 million savings, dependent upon block distance, with the \$0.396/L (\$1.50/gal) fuel.

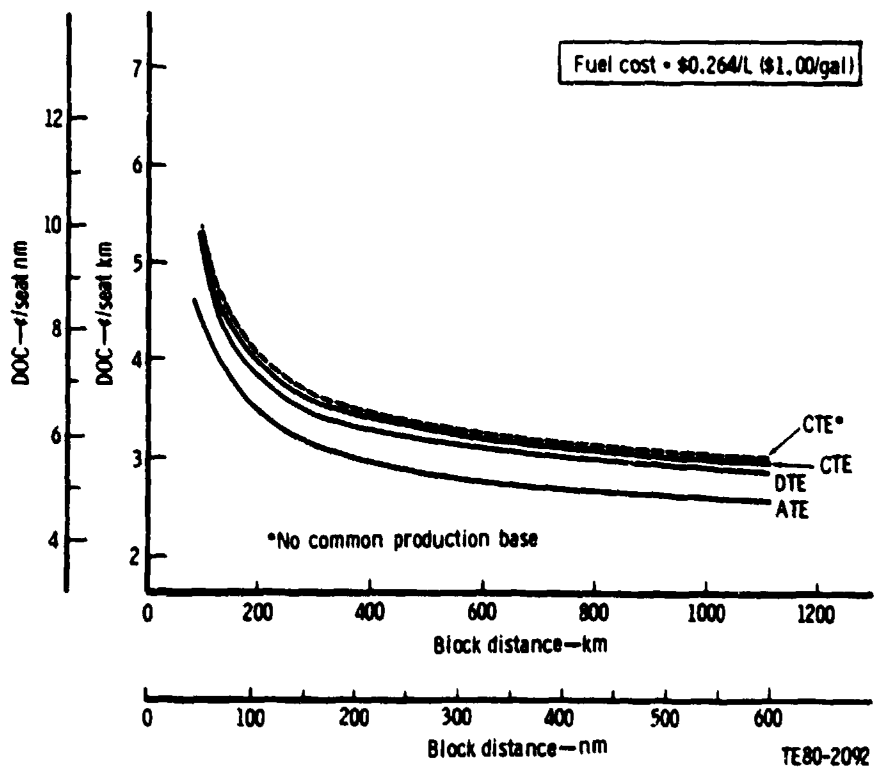


Figure 60. - DOC versus block distance--low speed aircraft, fuel cost \$0.264/L (\$1.00/gal).

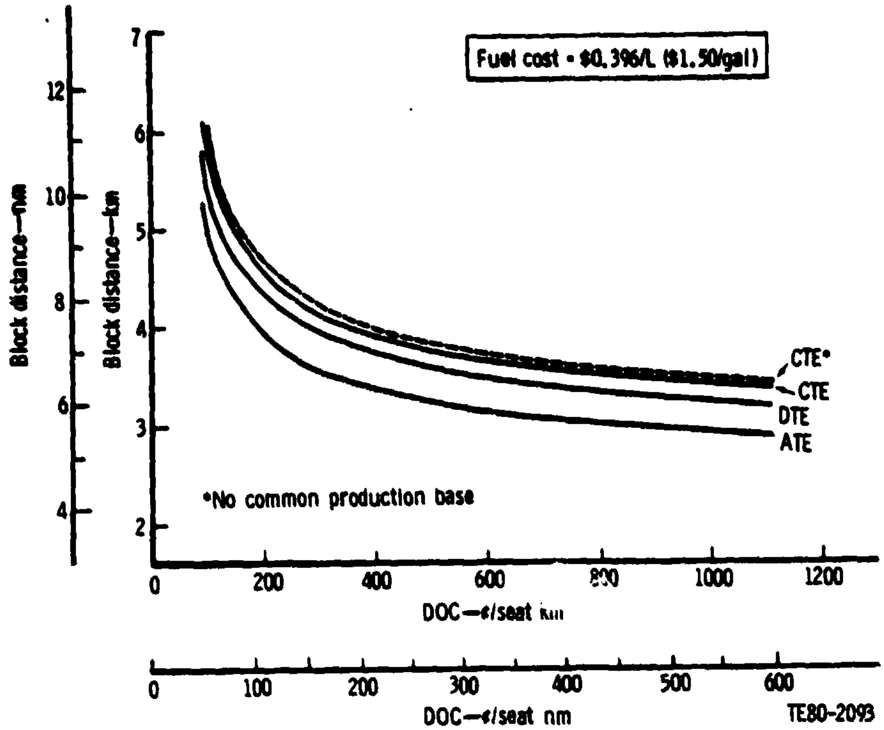


Figure 61. - DOC versus block distance--low speed aircraft, fuel cost $0.396/L ($1.50/gal).

The fuel burned, or fuel consumption improvements, obtained with the DTE and ATE are illustrated in Figure 67. This figure shows percent reduction in fuel consumption, relative to the CTE, plotted against block distance. The reductions for both the DTE and ATE are indicated to be essentially constant over the range of block distances, with the DTE obtaining an approximate 13% reduction, and the ATE an approximate 19.5% reduction in fuel burned per mission. The fuel savings that could be achieved over a 10-year period of operation with a fleet of 100 aircraft is shown in Figure 68. This figure plots DTE and ATE fuel savings, relative to the CTE, versus block distance. Figure 68 indicates relatively higher savings for the shorter block distances, with the DTE savings ranging from 322 to 284 million liters (85 to 75 million gal). The fuel savings obtained with the ATE range from 492 to 454 million liters (130 to 120 million gal) of fuel.

The variation in block speed across the range of block distance examined in this study is shown in Figure 69. This figure shows a change in block speed from 259 km/h (140 kt) for the 92.6 km (50 nm) distance to 482 km/h (260 kt) for the 1111.2 km (600 nm) design stage length, with the block speed for the 185.2 km (100 nm) distance indicated to be approximately 352 km/h (190 kt).

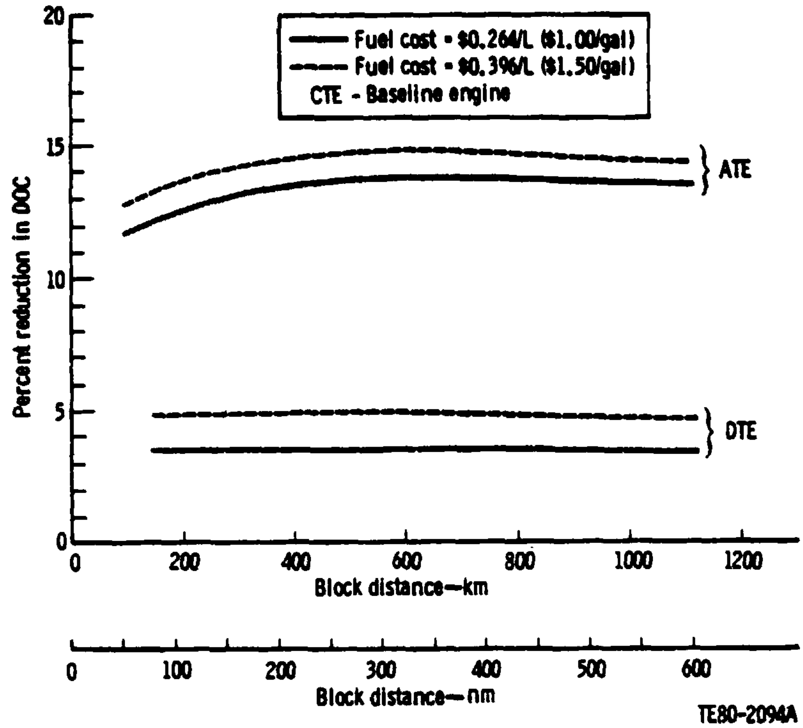


Figure 62. - DOC reductions—low speed aircraft.

A summary of the percentage changes in critical aircraft and cost parameters resulting from engine technology improvements is shown in Table LXVIII. These percentage changes are relative to the CTE-powered aircraft values. The cost parameters were calculated for the 185.2 km (100 nm) stage length mission (185.2 km (100 nm) block distance) only.

The combination of the sfc, weight, and envelope dimensions of the ATE is shown to have reduced aircraft TOGW by 4.5%. However, the total aircraft cost is shown to be essentially the same as the CTE-powered aircraft. This is because the lower airframe less power section cost was offset by the higher acquisition cost of the ATE compared to the CTE. It is noted that the significantly higher (with respect to the ATE) acquisition cost of the DTE contributed significantly to a 6% increase in total aircraft cost over the CTE.

The fuel consumption percentage changes basically reflect the sfc improvements associated with the DTE and ATE plus the rated engine power reduction associated with the gross weight reduction. The approximate 9% difference between the DOC and TCO reductions shown for the DTE and those shown for the ATE is a result of the combination of larger sfc, engine acquisition, and maintenance cost improvement associated with the ATE relative to the DTE. It is noted that increasing the fuel cost from $0.264 to $0.396/L ($1.00 to $1.50/gal) produces a 1% larger reduction in DOC and TCO for both the derivative and advanced engines compared to the current engine.

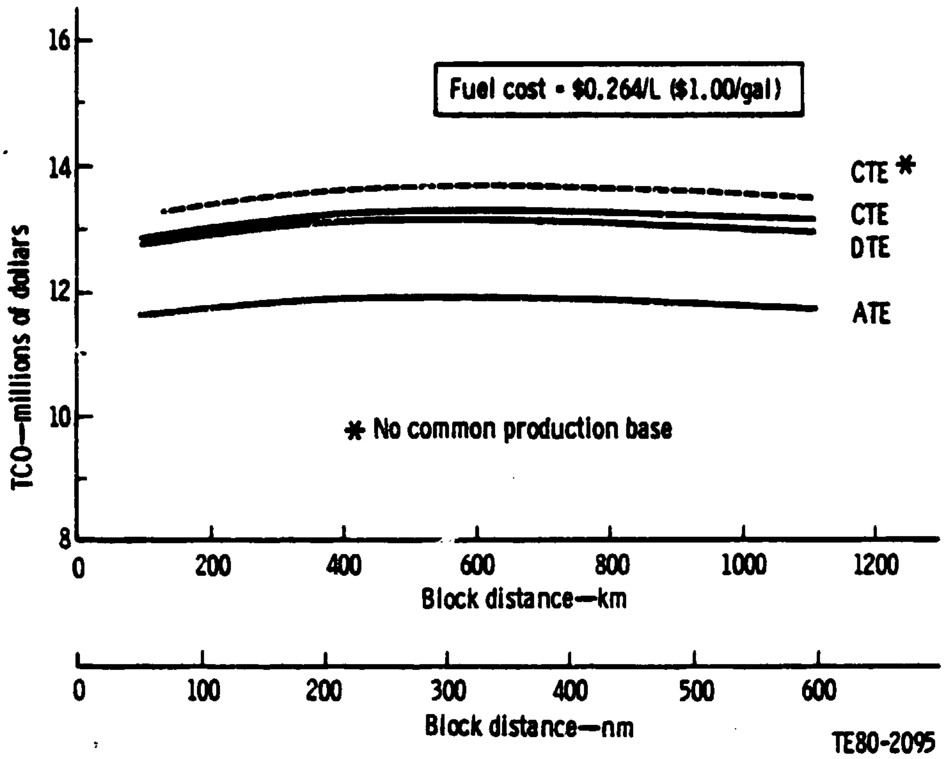


Figure 63. - 5-year TCO versus block distance--low speed aircraft, fuel cost $0.264/L ($1.00/gal).

TABLE LXVIII. - TECHNOLOGY COMPARISON--LOW SPEED AIRCRAFT

185.2 km (100 nm) block distance

		Percent change from CTE		
	CTE	'CTE'	DTE	ATE
TOGW	Base	0	-2.4	-4.5
TAC	Base	+6.0	+6.3	-0.7
Fuel consumption	Base	0	-12.8	-19.6
DOC at ①	Base	+1.6	-3.6	-12.6
DOC at ②	Base	-1.4	-4.9	-13.6
5-y TCO at ①	Base	+2.7	-1.2	-9.8
5-y TCO at ②	Base	+2.4	-2.6	-10.9

① fuel cost = $0.264/L ($1.00/gal)
② fuel cost = $0.396/L ($1.50/gal)
Utilization = 2800 h/y

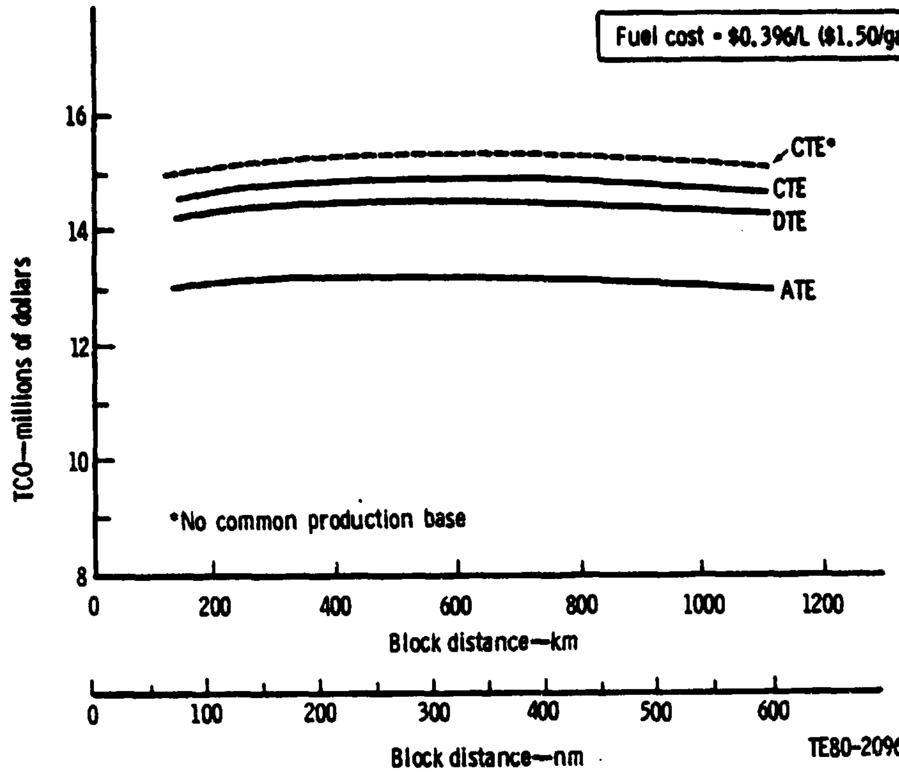


Figure 64. - 5-year TCO versus block distance--low speed aircraft, fuel cost $0.396/L ($1.50/gal).

A comparison of the 10-year fuel requirements of an assumed fleet of 100 aircraft flying the 185.2 km (100 nm) stage length exclusively is shown in the bar graph presented in Figure 70. This figure indicates a savings of 326 million liters (86 million gal) for the DTE and 496 million liters (131 million gal) of fuel for the ATE.

A comparison of the 10-year DOC for the fleet and mission prescribed in the foregoing fuel comparison is shown in Figure 71 for both $0.264 and $0.396/L ($1.00 and $1.50/gal) fuel costs. This bar graph indicates a savings on the order of $115 million for the DTE, and $316 million for the ATE at the $0.396/L ($1.50/gal) fuel cost.

A breakdown of the cost element in DOC for each of the technology engines is shown in Figure 72.

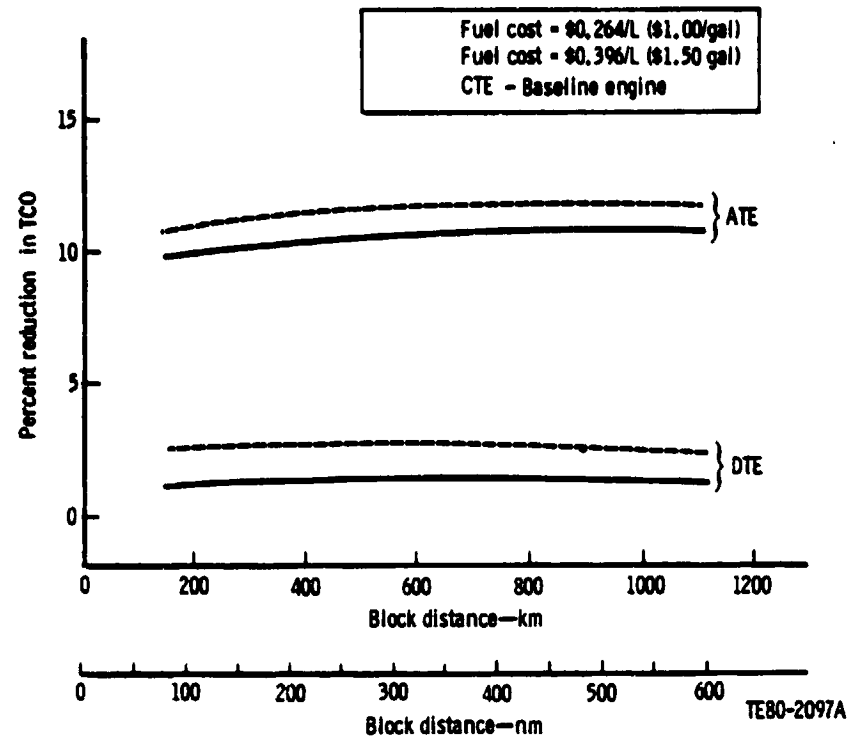


Figure 65. - 5-year TCO reductions--low speed aircraft.

This figure indicates that the DOC reduction for the DTE was essentially a result of the sfc improvement. Note that the sfc improvement was offset to a small extent by the higher depreciation and insurance costs associated with the higher engine acquisition cost. The DOC reduction noted for the ATE was essentially a result of a combination of significant sfc reduction, engine acquisition, and maintenance cost improvements.

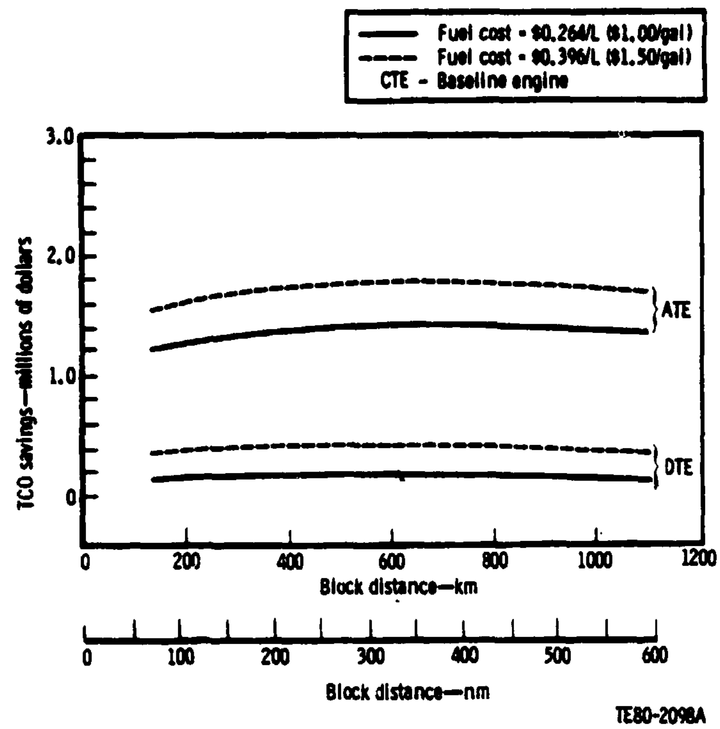


Figure 66. - 5-year savings--low speed aircraft.

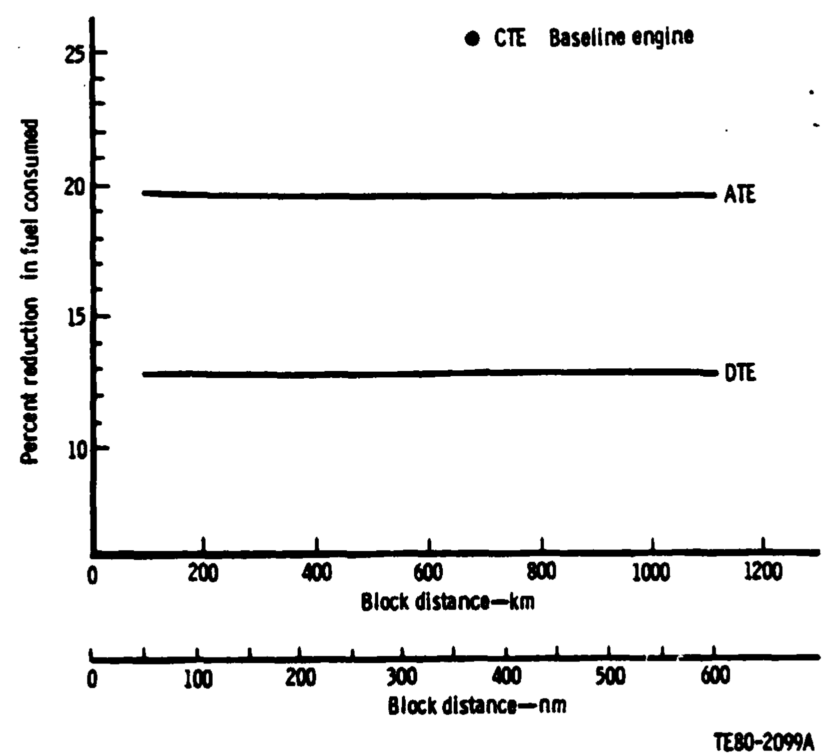


Figure 67. - Percent reduction in fuel consumed--low speed aircraft.

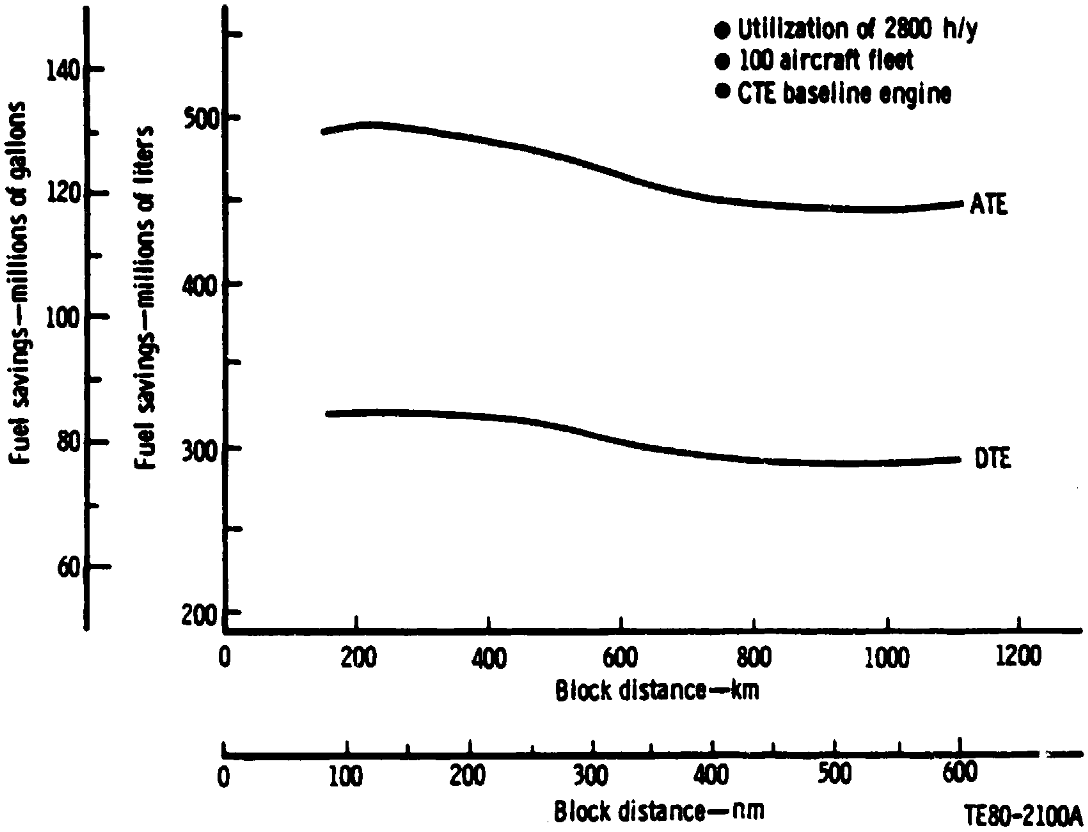


Figure 68. - 10-year fuel savings--low speed aircraft.

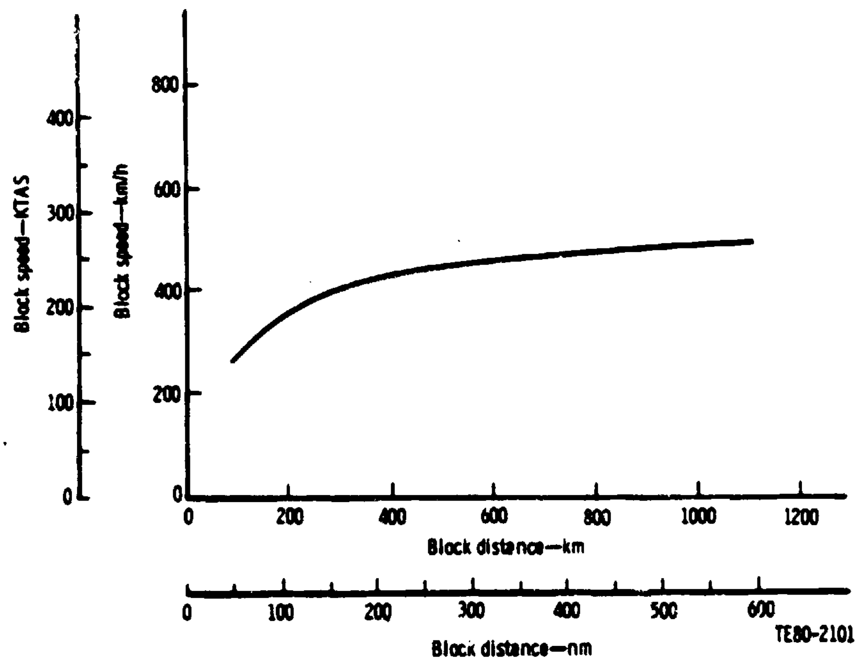


Figure 69. - Block speed versus block distance--low speed aircraft.

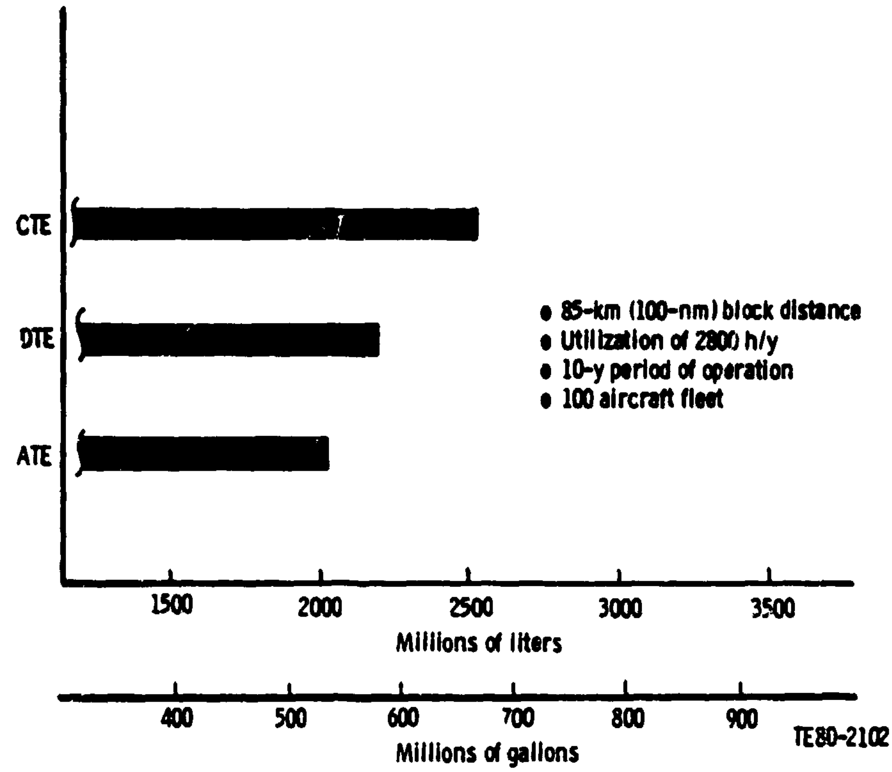


Figure 70. - Fuel consumption comparison--low speed aircraft.

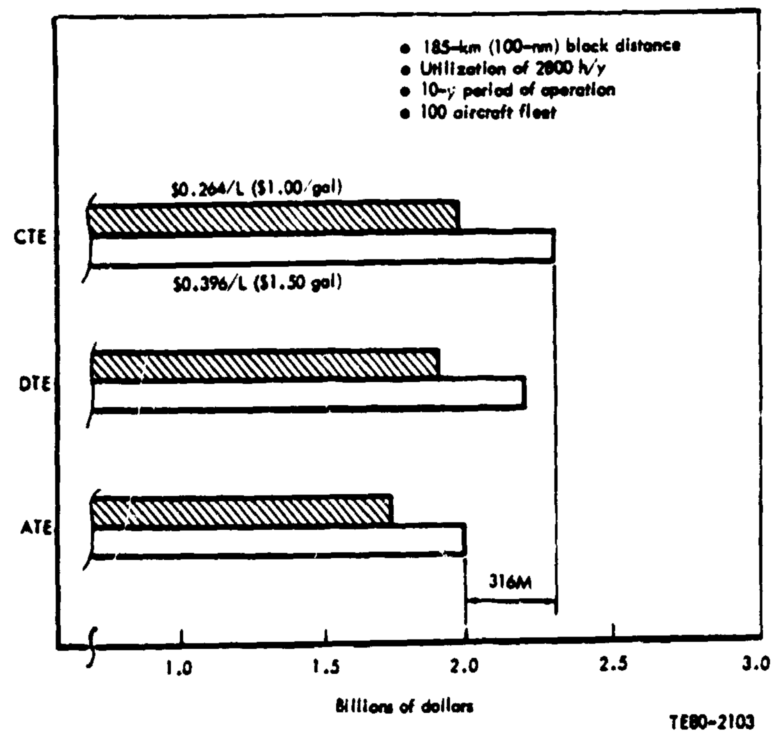


Figure 71. - DOC comparison--low speed aircraft.

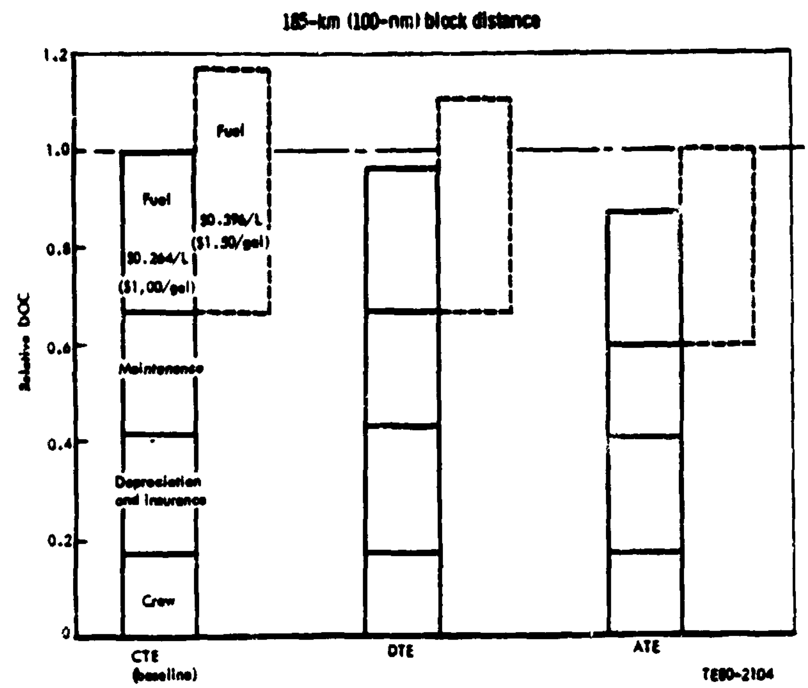


Figure 72. - DOC breakdown--low speed aircraft.

RECOMMENDATIONS FOR FUTURE RESEARCH

NASA should address basic research and development needs for the near term to broaden the industry data base for design of advanced gas turbine engines for small transport aircraft. A general program directed toward specific areas where an investment in R&D dollars could produce critical data needed in design of new turboprop engines for commuter application will be presented. These results are based on a generalized preliminary design study of commuter aircraft turboprops at 1790 kw (2400 shp) and 3579 kw (4800 shp) sizes for two different 50-passenger aircraft differing principally in initial cruise altitude and design Mach number.

As advanced transport needs are resolved through STAT vehicle studies and user requirements, a useful purpose could be fulfilled by implementing an experimental engine program in a size consistent with 1990 projections of vehicle requirements. Although most new technology voids can be filled by component and material programs, the application of certain high risk design innovations must be assessed by experiment to develop sufficient confidence to proceed with full development.

PROGRAM CONTENT

The overall program content is shown in Figure 73. This program includes the STAT propulsion study, basic R&D effort, and component R&D in the area of compressors, turbines and shafting, advanced structures, combustors, and controls and systems, leading to an experimental engine program that includes a design study, further component tests, test of a gas generator core, and finally the experimental core with the power turbine and propeller gearbox added.

The experimental engine program is a four-year program led by a design effort in which results of the component R&D efforts are integrated into the experimental engine design. About 200 hours of developmental engine testing are included in the program.

The experimental engine provides a means to continue basic R&D on high risk components in an engine environment, and also provides a vehicle to gather data on high risk mechanical systems such as a full-time turbine active clearance control. This is an example of a system that would not be released for development on an engine program without some experimental engine experience.

Certain undeveloped technology elements have been identified during this study program as essential in the full-scale development of the STAT engines. Research programs leading to the fulfillment of these element requirements are described herein, and are listed in Figure 74.

BASIC RESEARCH AND DEVELOPMENT

STAT studies show that the largest improvements in commuter aircraft DOC, as influenced by the engine, are achieved through reduced engine fuel consumption and reduced maintenance cost.

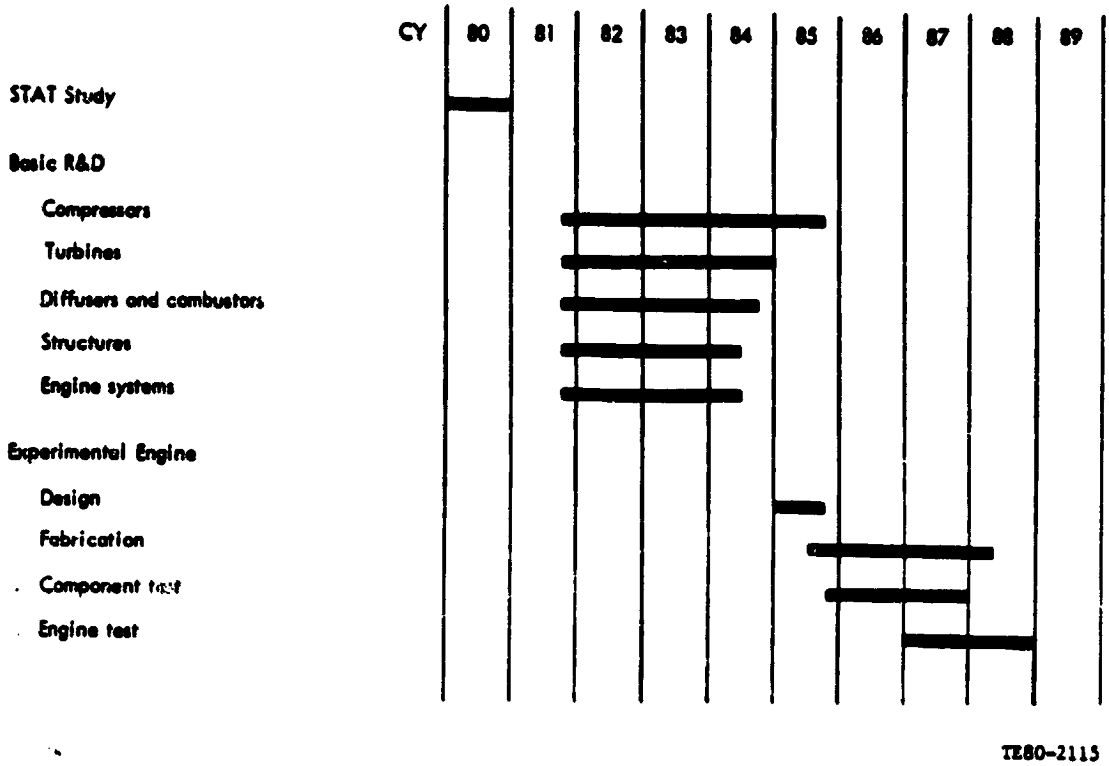


Figure 73. - STAT advanced technology program.

Fuel consumption can be reduced through improved thermodynamic cycle efficiency. Compressor pressure ratios around 20:1 and turbine RIT's near 1506 K (2250°F) appear to offer the minimum DOC. Advanced engine component physical size is greatly reduced compared to current technology, which brings about a new set of challenges to realize the potential efficiency gains, and opportunities to reduce engine cost through application of new technology. As engine pressure ratio and RIT increases, more emphasis must be placed on development of technology applcable to small high pressure ratio compressors of various types including axial and axial-centrifugal, both single and dual spool. New turbines, featuring advanced construction methods and new materials, are needed to improve cooling techniques and permit efficient blade design in the relatively small flow passages available. Bearing and shafting technology must be advanced to permit increased rotative speeds with improved dynamics and increased bearing life. Engine reliability and maintainability must be improved to achieve low maintenance costs and on-time performance. Basic research and development programs which are needed to achieve these gains are presented on Table LXIX with the benefit/cost ratio, rank, and probability of success shown for each program. Benefits are based on the reduction in total DOC assignable to each technology as applied to a fleet of 100 aircraft powered by current technology engines, operating at 2800 hr per year for a 10-year period on a typical route segment of 100 nautical miles and using fuel priced at $0.264/L ($1.00/gal).

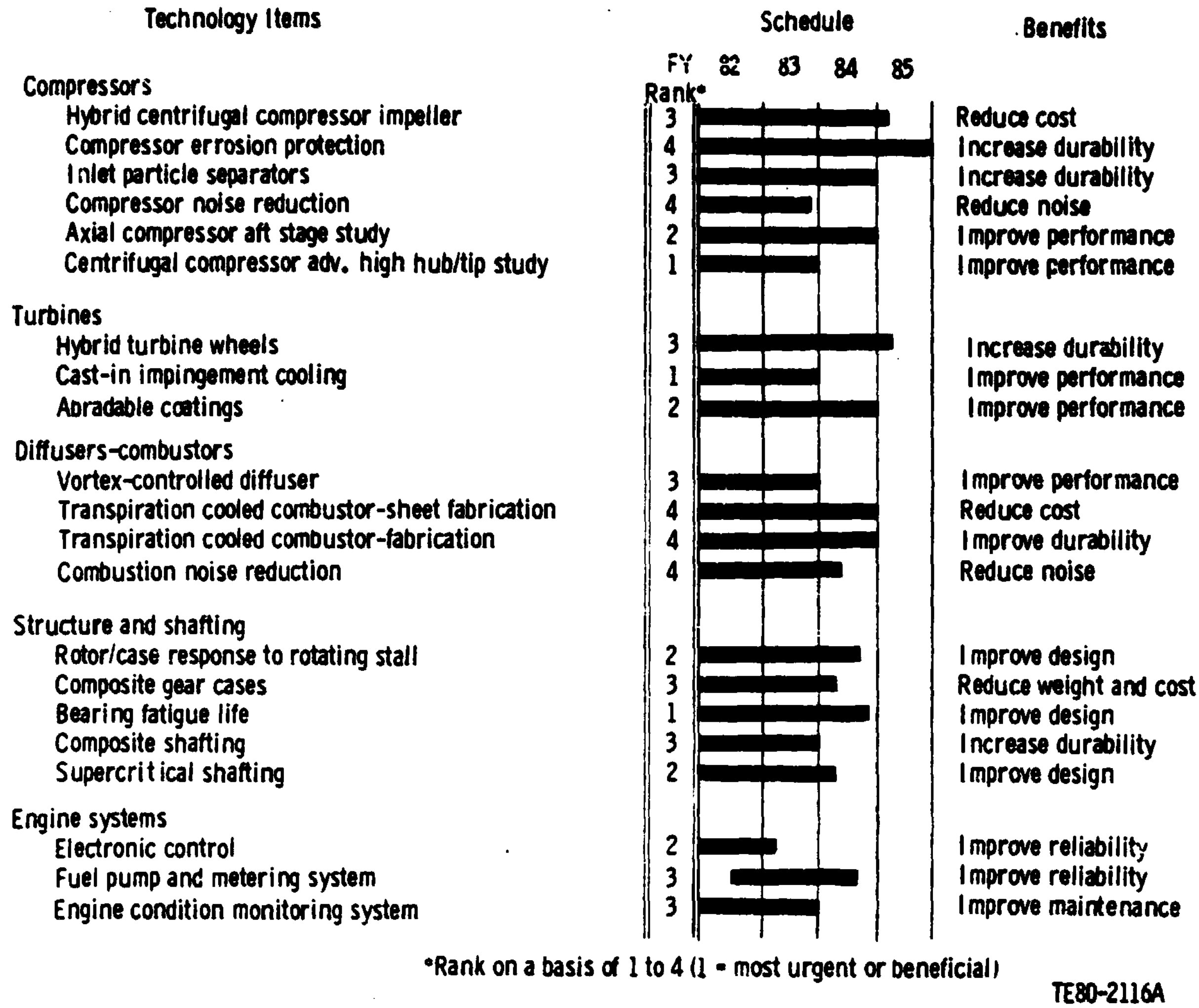


Figure 74. - STAT technology research programs.

Compressors

Compressors chosen for STAT engines feature high pressure ratio, single-spool configurations with reduced numbers of stages and reduced blade count. Split-spool configurations appear to offer similar gains in performance with generally increasing complexity and somewhat lower risk.

Axial-centrifugal compressors tend to result in shorter engines when used with foldback combustors. This, in turn, reduces shaft length and eases dynamic problems at the expense of greater diameter at the engine midsection and probably lower overall compressor efficiency. Axial compressors tend toward higher efficiency; however, small blade sizes in the latter stages can result in performance penalties that may offset fundamental gains. An expanded compressor data base is required to make the proper choice in advanced engine design.

TABLE LXIX. - CRITICAL TECHNOLOGY ELEMENTS FOR STAT RESEARCH AND DEVELOPMENT

Engine Technology Program	Cost $000's	Benefit $000,000's	Benefit/ Cost Ratio	Rank*	Probability of Success
Power Section					
o Compressor					
Hybrid Cent. Compr. Impeller	560	5.851	10.45	12	Likely
Axial Compr. Aft Stage Study	620	16.046	25.88	5	Likely
Cent. Compr. Adv. Hi-Hub/Tip Study	680	7.263	10.68	11	Likely
Compr. Erosion Protection	620	10.503	16.94	8	Likely
Inlet Particle Separators	500	12.253	24.51	6	Likely
Rotor/Case Response to Rotat, Stall	300	15.170	50.57	2	
o Turbine					
Hybrid Rotors	560	1.459	2.61	19	Likely
Composite Shafting	400	2.042	5.11	17	Likely
Super Critical Shafting	350	1.750	5.00	18	Likely
Cast-In Impingement Cooling	250	1.750	7.00	16	Likely
Abradable Coatings	680	13.128	19.31	7	Likely
Bearing Fatigue Life	280	2.917	10.42	13	Assured
o Combustor/Diffuser					
Vortex-Controlled Diffuser	490	8.169	16.67	9	50/50
Transpiration-Cooled Comb,-Sheet Fab	530	0.875	1.65	20	Likely
Transpiration-Cooled Comb, Fab	650	0.583	0.90	21	Likely
o Engine Accessoried					
Electronic Fuel Control	250	2.917	11.67	10	Likely
Fuel Pump & Metering System	400	3.501	8.75	14	Likely
Engine Condition Monitoring	400	29.174	72.94	1	Likely
o Noise Reduction					
Compressor Noise Reduction	250	6.710	26.84	4	Likely
Combustor Noise Reduction	500	4.084	8.17	15	Likely
Reduction Gear					
o Composite Gear Case	300	11.670	38.90	3	Likely

*Ranked on basis of Benefit/Cost Ratio

In order to reduce maintenance cost, a primary STAT engine requirement, it is beneficial to protect the engines from dust-laden air common to many airports used by commuter transports. This protection may be obtained by filtering dust particles from the inlet air, making the compressor more tolerant of the dirty air, or a combination of both.

The state of the art for advanced propellers and propfans is such that they will exhibit much lower noise generation than do current propulsors. For this reason, it is essential that advance turboprop engines be considerably quieter than current engines of comparable power. The STAT engines, in particular, due to their high cycle pressure ratios and resulting supersonic compressor blade tip velocities, require some form of compressor noise reduction.

The high pressure ratio of the STAT engines is achieved with compressors which are designed with minimum rotor tip clearances and have highly loaded stages, compared with current technology engines. One of the problems encountered in designing such compressors is predicting tip clearance during periods of dynamic structural response due to surge, rapid thermal gradients, and the relatively unknown phenomenom of rotating stall.

The small diameter, high speed requirement of the 1790 kw (2400 shp) STAT engine presents a problem for the design of a practical compressor impeller. The bore loads for such an impeller preclude the use of a conventional cast impeller. The solution selected for the STAT engine is the use of an advanced hybrid impeller with a forged hub diffusion bonded to a cast outer shell with blades.

Compressor research programs recommended are:
- o Hybrid centrifugal compressor impeller
- o Compressor erosion protection
- o Inlet particle separators
- o Axial compressor aft stage study
- o Centrifugal compressor-advanced high hub/tip ratio study
- o Rotor/case response to rotating stall

Hybrid Centrifugal Compressor Impellers

The STAT 1790 kw (2400 shp) engine incorporates a centrifugal compressor impeller in its last stage. The high speed and small size of this impeller is such that a conventional cast impeller will not meet the bore loading requirements. It is necessary to develop a hybrid impeller for this application, which has a high strength, forged hub diffusion bonded to a cast outer shell with blades. An example of this type of impeller is shown in Figure 75.

This STAT program will address to the heat treat response of selected alloy combinations, the generation of a data base for a single alloy combination and component cyclic spin (LCF) testing.

Compressor Erosion Protection

The STAT engine requirement for minimum maintenance cost will be met in part by designing its compressor to be tolerant of dirt particles ingested in the airstream, as shown in Figure 76. A need exists for a computer model that will enable the designer to predict the dirt tolerance of any given compressor configuration, and thereby enable him to select the optimum design from several candidates.

The elements required to develop such a computer model include:

- o Modification of an existing solid particle trajectory calculation model to account for particle impacts with the rotating and stationary blading within a compressor.
- o A new mathematical model for the mechanics of-
  - Particle rebound from compressor surfaces
  - Particle shattering upon impact
  - Compressor blading and side-wall material loss
- o Determination of performance and stability of an eroded compressor from the effects of-
  - Clearance changes
  - Increased roughness
  - Decreased solidity
  - Blade shape change

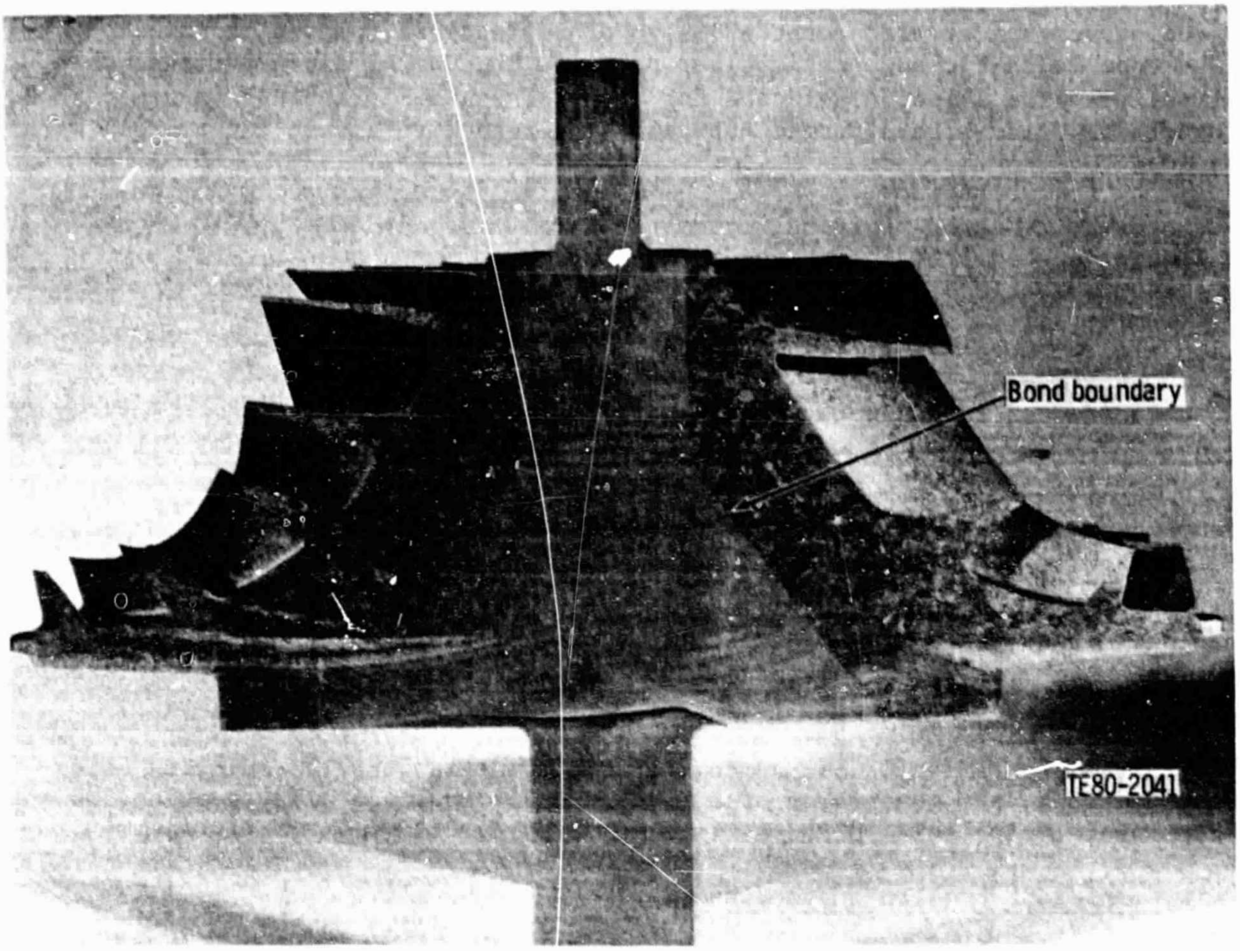


Figure 75. - Dual-property titanium impeller with a wrought Ti-6246 hub HIP bonded to a cast Ti-6242 airfoil shell.

- o Test validation--A test program will be conducted with controlled contaminant ingestion to ascertain compressor performance loss, stability loss, and erosion damage resulting from contaminant ingestion. The test program will include two single-stage test compressors, one representing current technology and the other designed to minimize erosion effects.

Inlet Particle Separators

In meeting the STAT requirement for minimum maintenance cost, one of the areas of concern is preventing ingestion of dirt-laden air, which is the primary cause of erosion damage causing performance loss and frequent repair. An inlet particle separator (IPS) may be required for the STAT engines to eliminate this problem. Basic test data and analytical studies are required to design an efficient and cost-effective IPS for the STAT engine commuter transport aircraft.

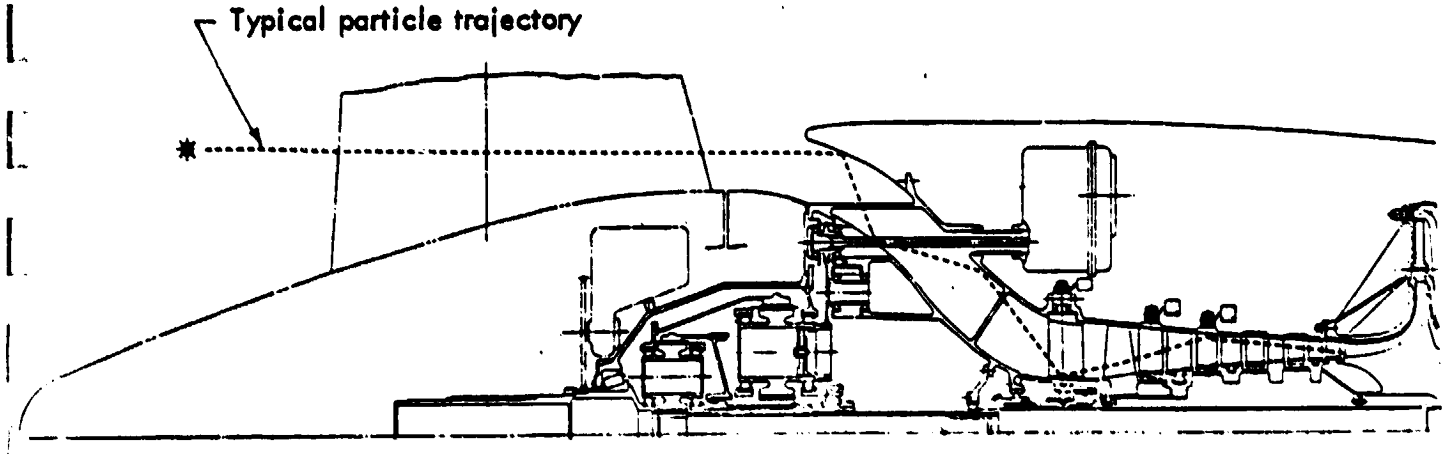


Figure 76. - Foreign particle ingestion phenomenon.

An existing particle trajectory calculation will be used to assess the potential separation efficiency of various IPS concepts suitable for use with STAT engines. Since the calculation is for axisymmetric or two-dimensional ducts, the nonaxisymmetric cases will be modeled as piecewise two-dimensional or axisymmetric.

The two most promising candidate concepts (one axisymmetric and one nonaxisymmetric) will be selected for detailed design and testing on a flow rig. A typical concept is shown on Figure 77. The testing will allow determination of pressure drop through the separation devices, inlet total pressure distortion at the exit of the device (which is the compressor inlet plane), and separation efficiencies of the device for various types of test contaminants such as aircraft coarse dust and Mil-C-sand.

This STAT research program will provide the basic data required for the development of a practical IPS for the STAT engines.

Rotor/Case Response to Rotating Stall

Increased performance and reduced life cycle costs desired for compressors of advanced gas turbine engines, like the STAT engines, are being achieved by designs with fewer and more highly loaded compressor stages, while maintaining tight rotor tip clearances. This simultaneous achievement of higher aerodynamic loading and minimum tip clearance is extremely difficult because of the dynamic behavior of the rotor and case during surge, rotating stall (Figure 78), and periods of rapid thermal gradients.

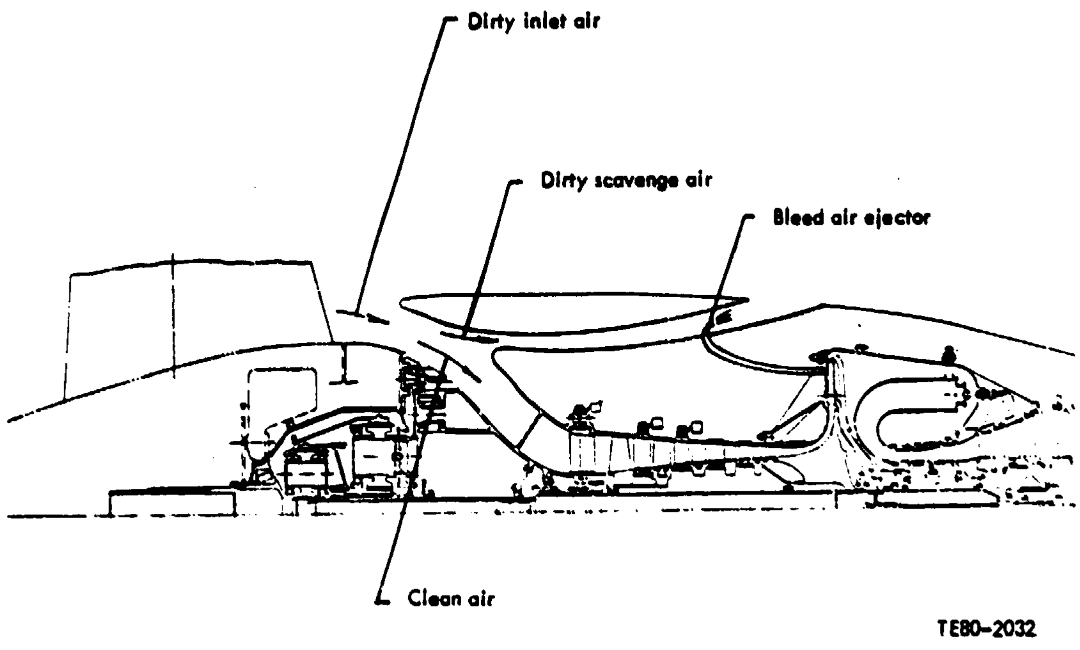


Figure 77. - Inlet particle separator concept.

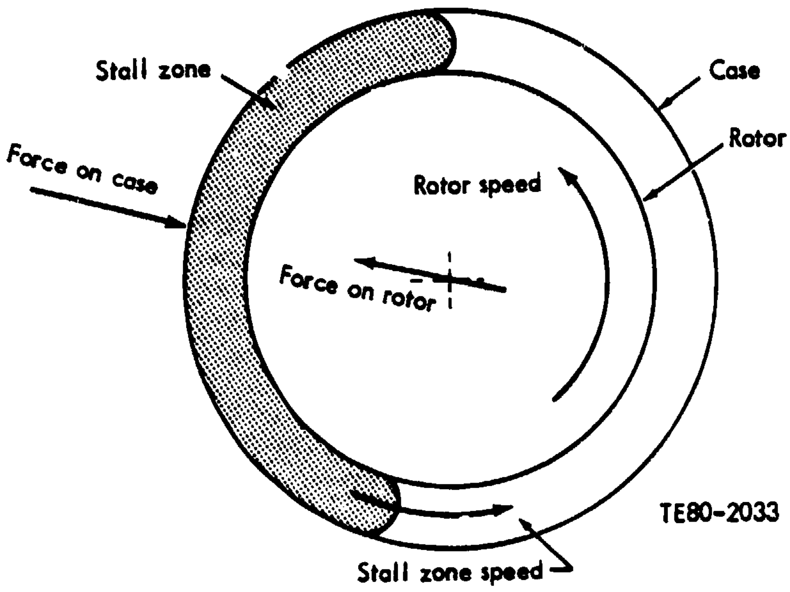


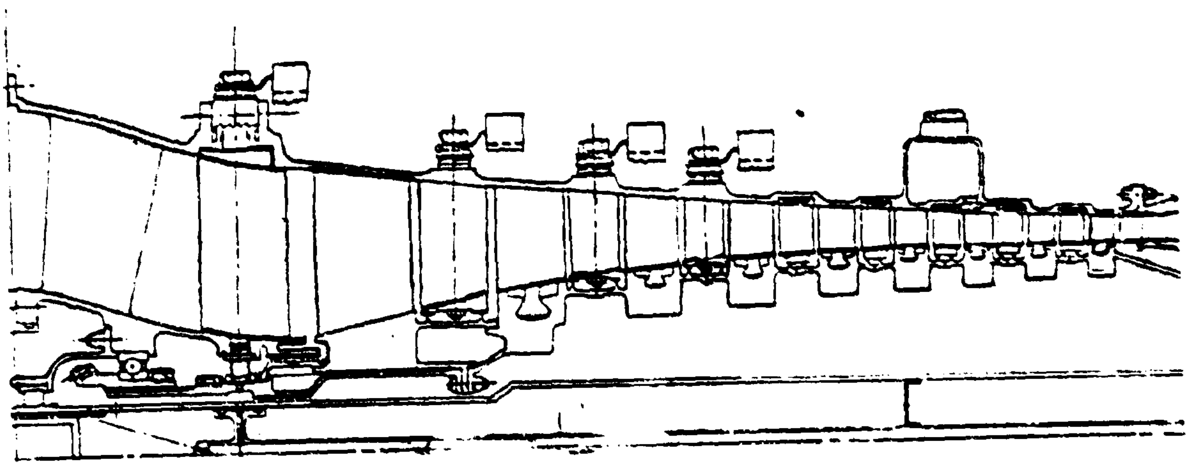
Figure 78. - Rotating stall phenomenon.

A predictive capability which will yield accurate information regarding tip clearance variation during periods of dynamic structural response is vital to the success of the STAT advanced compressor designs. Current analytical techniques do not account for asymmetric effects in rotor/case coupled, structural response. These effects, due to such common features as compressor horizontal flanges or bleed manifolds, can be very important in the structural dynamic response of compressors.

This STAT research program would provide an analytical capability recognizing these asymmetric effects, and thus yield an improved predictive capability compared to the current state of the art. Additionally, this program will yield experimental data describing the structural response of an advanced turboshaft compressor.

Axial Compressor Aft Stage Study

The STAT engines have a high pressure ratio, compared to that of current technology engines, in order to achieve the desired fuel saving performance. This NASA research study would be concerned with potential problems associated with the very small blade rows comprising the latter stages of high pressure ratio/low flow axial compressors, as shown in Figure 79. To improve the erosion resistant characteristics of these small airfoils, nonoptimum distributions of thickness-to-chord and leading edge radius will be needed. A systematic analytical/experimental investigation into the performance characteristics of these nonoptimum airfoil sections will add credance to performance estimates of potential STAT compressors. A determination of the sensitivity of these small blade rows to production tolerances including fillet geometry, angle variation, and surface finish, is needed. The compromises between close tolerance (higher manufacturing cost) and aerodynamic performance with the resulting effect on DOC can be more accurately assessed with the data from this program. Reduced sensitivity of these stages to tip clearance will be a third part of the program work plan. Possible aerodynamic improvements may include blade track trenching and low loss end-wall loading distributions.



TE80-2039

Figure 79. - Axial compressor.

Centrifugal Compressor-Advanced High Hub/Tip Ratio Study

Derivation of the optimum cycle performance for the 1790 kW (2400 shp) STAT engine resulted in the selection of an axial-centrifugal compressor as described earlier in this report. Designing the centrifugal stage for this compressor under all operating conditions will require certain data not currently available.

This NASA research study would deal with problems associated with designing high hub/tip ratio centrifugal compressors to operate behind an axial compressor, as shown in Figure 80. In general, the centrifugal stage of an axial-centrifugal compressor will be characterized by high inducer hub/tip radius ratio and low specific speed. Operating behind an axial compressor, the inducer would have to accept distorted inlet conditions, which would vary with speed and loading. The high inducer hub/tip ratio results from geometric matching to the axial compressor and the desire to reduce engine length (reduced transition from axial exit to centrifugal inlet). The low specific speed (i.e., reduced mechanical rotative speed for a given flow and pressure ratio) results from tradeoffs between axial and centrifugal compressor efficiencies.

The combination of high hub/tip ratio and low specific speed results in an impeller flow path that looks quite different from most modern high performance centrifugal compressors, which are designed to the lower hub/tip ratios and higher specific speeds to maximize performance potential. It is not surprising, then, that the bulk of available data defining state-of-the-art centrifugal performance is based on these "more optimum" designs.

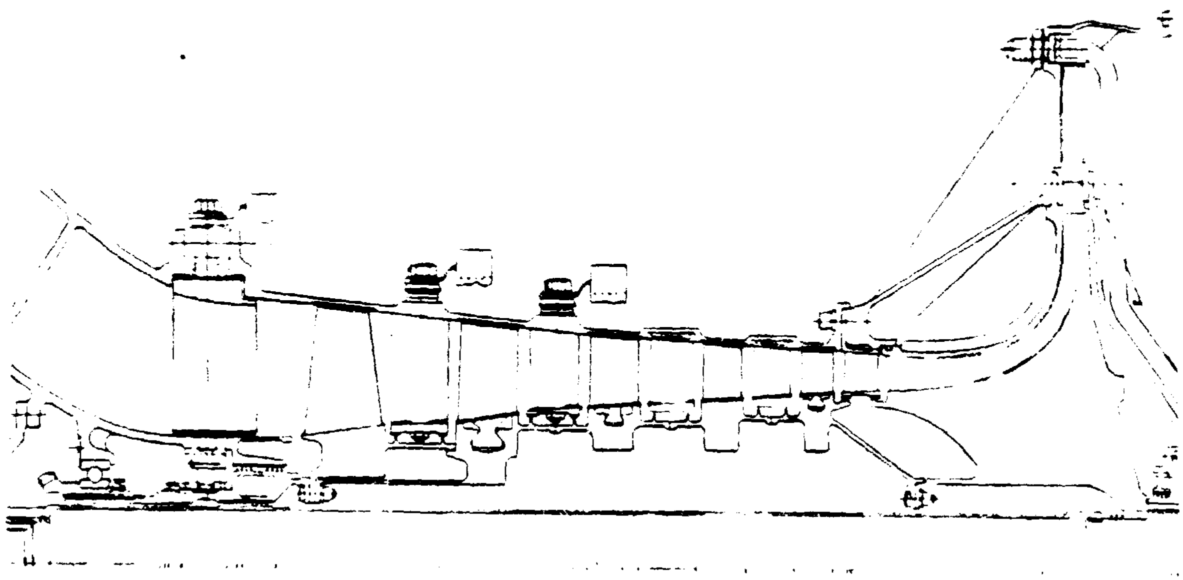


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Figure 80. - Axial-centrifugal compressor.

High impeller exit back curvature is generally accepted as having positive benefits on compressor performance, and will probably be incorporated in advanced axial-centrifugal designs. The effects of high inducer hub/tip ratio on stage performance of these advanced compressors will be needed for future configuration studies.

Dual-stage centrifugal compressor testing has shown reduced performance levels for the second stages. These reduced performance levels are currently attributed to distorted inlet conditions being delivered by the intrastage crossover duct. Similar inlet effects would be expected for operation behind an axial compressor. The proposed High Hub/Tip Centrifugal Compressor Program would address the inlet effects by testing with "clean" and "distorted" inlet conditions.

Turbines

With the higher pressure ratio and higher turbine temperature chosen for the STAT engine cycle, the gasifier spool speed increases, and its diameter decreases, relative to current technology engines. Turbine configurations chosen for STAT engines feature hybrid turbine wheels that can meet the resulting requirement for bore load-carrying capability within the size contraint.

The small size of the turbine wheels makes it virtually impossible to use individual turbine blades with conventional dovetail attachment. The hybrid turbine wheels provide the solution to this problem by incorporating blades cast onto a ring, which, in turn, is diffusion bonded to the wheel hub. Also, the small size of the turbine blades make necessary the use of cast-in impingement cooling air passages.

The small size and high speed of the main rotor shafts in the STAT engines make it difficult to design the main bearings with satisfactory load-carrying capacity and fatigue life. Considerable promise of improving these characteristics exists in better understanding the formation and control of forging flow lines in the bearing balls.

The small diameter of the STAT engines makes it impractical to use subcritical, conventional steel shafting. Metal matrix composites, however, offer improved mechanical properties and can meet the desired shaft strength and stiffness within the size constraints. As a backup to composite shafting (should that material development not be achieved in the STAT program time frame), supercritical steel shafting offers an alternate "second" choice. Basic design information is required in the ability to predict dynamic response, and to control vibrations in supercritical shafts in order to obtain their benefit the the STAT engines.

Another area in which the small size of the advance technology STAT turbines presents a challenge is in maintaining minimum acceptable interstage leakage with practical production tolerances. The solution selected for the STAT turbines is the application of abrasive blade tip coatings and abradable tip seal coatings.

The turbine research programs recommended are:

- o Hybrid rotors
- o Composite shafting
- o Supercritical shafting
- o Cast-in impingement cooling
- o Abradable coatings
- o Bearing fatigue life

Hybrid Rotors

The small diameter of the STAT gasifier turbine, coupled with its high speed, makes impractical the use of separate turbine blades with conventional dovetail attachment. It is necessary to use a hybrid turbine wheel with blades cast onto a ring as one piece, and the ring, in turn is diffusion bonded to a high strength powder metallurgy hub.

The feasibility of using hybrid, or dual-property, turbine wheels in gas turbine design has been demonstrated in several previous programs at DDA. An example of such a turbine wheel is shown in Figure 81.

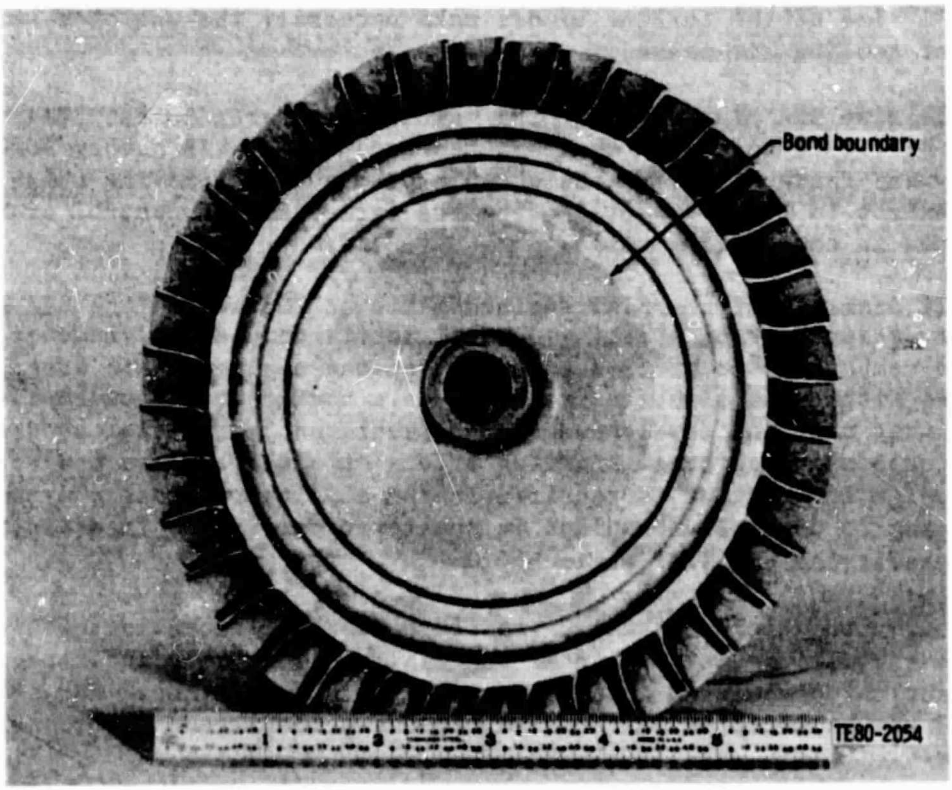

Figure 81. - Dual-property turbine wheel with PA-101 hub and Mar-M247 internally cast airfoil ring.

In all of these previous programs, however, the alloy combinations used were PA-101 hub and Mar-M247 airfoils, and little effort was allocated to bore load-carrying improvements.

In the proposed STAT research effort, alternate hub alloys (i.e., MERL76, AF95, AF115) will be evaluated for compatibility with Mar-M247. These alloys, along with a baseline PA-101 composition, will then be evaluated for post-HIP bonding response. Heat treatments with rapid cooling rates from the bonding and airfoil-coating diffusion cycle will be emphasized.

With establishment of preliminary heat treat response, a single alloy will be selected and heat treatments further refined. Following establishment of a fixed process, additional rotors will be produced, a mechanical property data base generated, and cyclic spin tests performed to evaluate LCF capabilities. The resulting data will enable the selection of materials for the STAT engines, which would provide an appreciably longer fatigue life than those of today's gas turbine engines.

Cast-In Impingement Turbine Cooling

The small size of the turbine blades on the STAT engines makes impractical the use of conventional separate impingement tubes through the blades for the delivery of blade cooling air. The obvious solution is to cast the cooling air supply passages into the blade as it is cast. The technique for this process is not currently available on a cost-effective basis.

This program would explore two possible core fabrication techniques in an effort to develop a reliable, cost-effective foundry process for producing air-cooled turbine airfoils with cast-in impingement tubes:

- o Assembled core
- o Fabricated core insert

The assembled core approach would be to inject the feed cavity core (marked A in Figure 82) and the collection cavity core (B in Figure 82) separately. The cores would then be assembled by inserting quartz rod (marked C in Figure 82) into holes formed in the cores during the injection process. "gill" discharge holes (marked D in Figure 82), or other film cooling holes, would be formed with quartz rods inserted through the wax shell prior to ceramic mold dipping.

The fabricated core insert approach involves the placement of an insert with final blade internal geometry into the core die prior to injection. The core material would then be injected around this insert forming a one-piece core with the insert embedded within it; the core would then be processed to remove the insert. Subsequent processing would then be similar to present blade ring fabrication.

The principal problem anticipated for the development of cast-in impingement cooling would be core fragility. Single casting trials could be made prior to core design in order to establish hole diameter and wall thickness constraints.

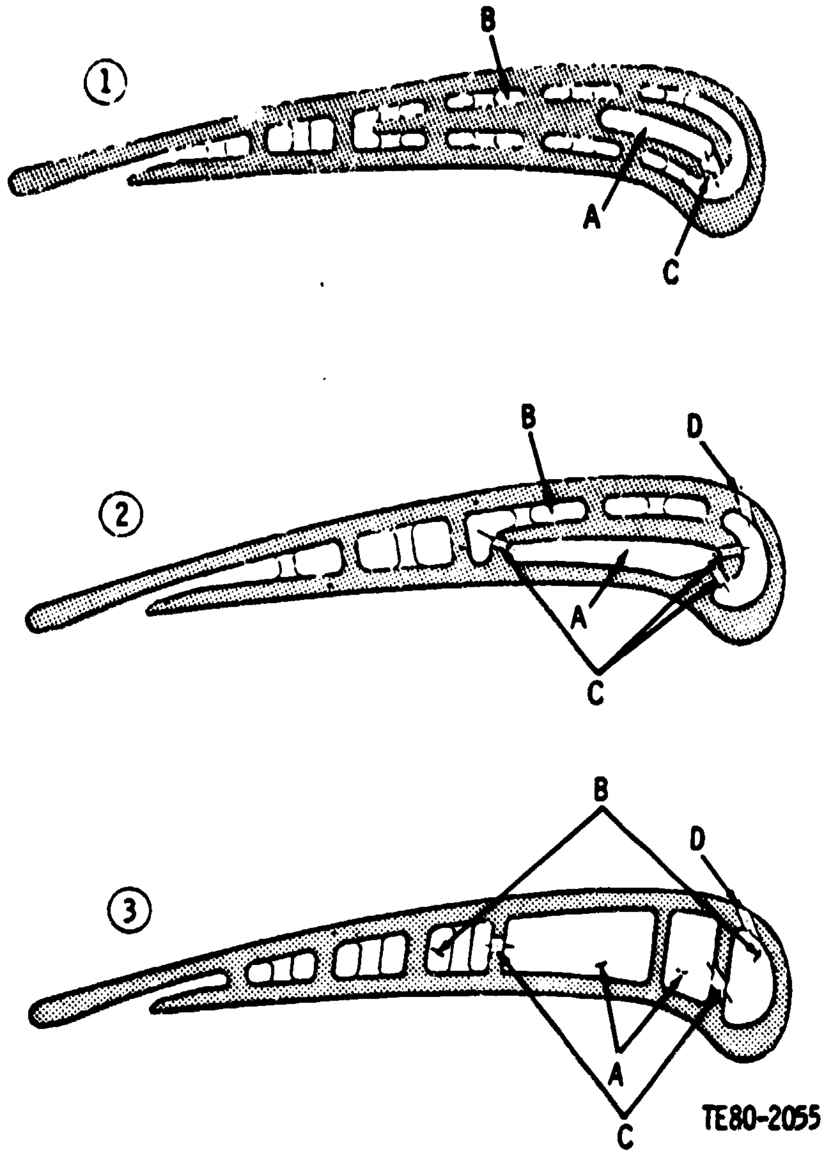

Figure 82. - Schematics illustrating candidate cast airfoil cooling schemes.

Abradable Coatings

The small diameter of the STAT engines presents a problem in maintaining acceptable interstage turbine leakage with practical production tolerances. To combat this problem, the STAT engines will use abradable surfaces on the tips of the turbine blades. In order to select the best material for this use and the process for its application, an investigative program is required.

This program will result in the development of the materials and processes required for direct application of abrasive particles to turbine blade tips, as shown in Figure 83. The abrasive elements will provide an augmented abradability capability for enhanced gas path sealing and improved cycle efficiency. A primary advantage of the direct application process is its applicability and affordability for small turbine rotors with integral blades. This concept has been taken satisfactorily through the proof-of-principle demonstration stage, and represents a development effort with reasonable technical risk.

Composite Shafting

The high pressure ratio of the STAT engines dictates high rotor speeds and small engine diameters, as compared to current technology engines. This situation compounds the problem of designing shafts to transmit torque from the power turbine through the gasifier rotor to the output shaft. It becomes impossible to use conventional, subcritical, forged steel shafts for such applications.

The best way to overcome the problem appears to be in the use of composite shafting, such as shown in Figure 84, which offers greatly improved mechanical properties over steel shafts. Composite shafts offer payoffs in the areas of engine weight and complexity on the basis of having available to the designer the potentially high E/ρ (Young's Modulus to density ratio), σ/ρ (tensile stress to density ratio), and τ/ρ (shear stress to density ratio) engineering properties. Before composite shafts can be successfully applied to a commercial engine, however, the difficulty of manufacturing such a shaft with accompanying biased-ply layups and end fittings, and of achieving the assumed high goals of engineering properties, must be overcome.

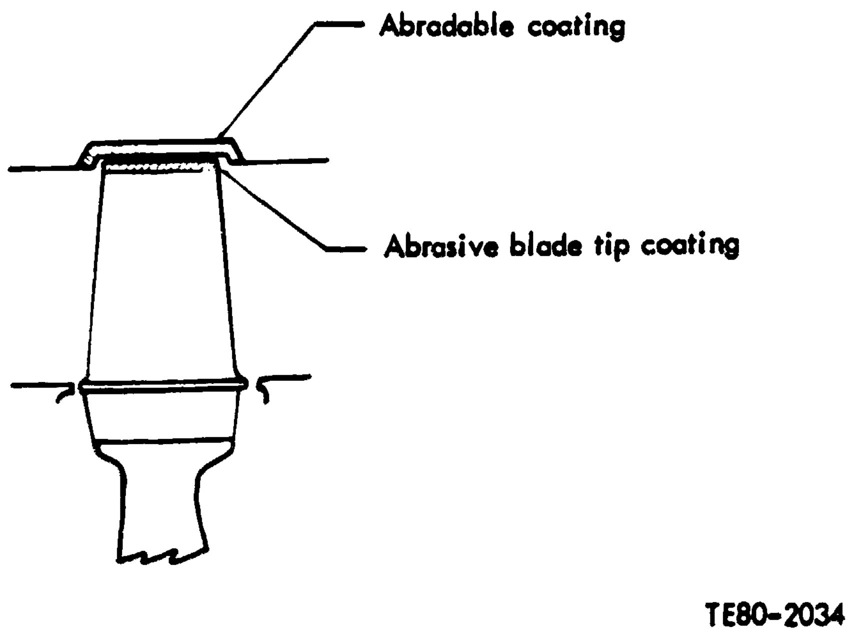

Figure 83. - High temperature turbine seal concept.

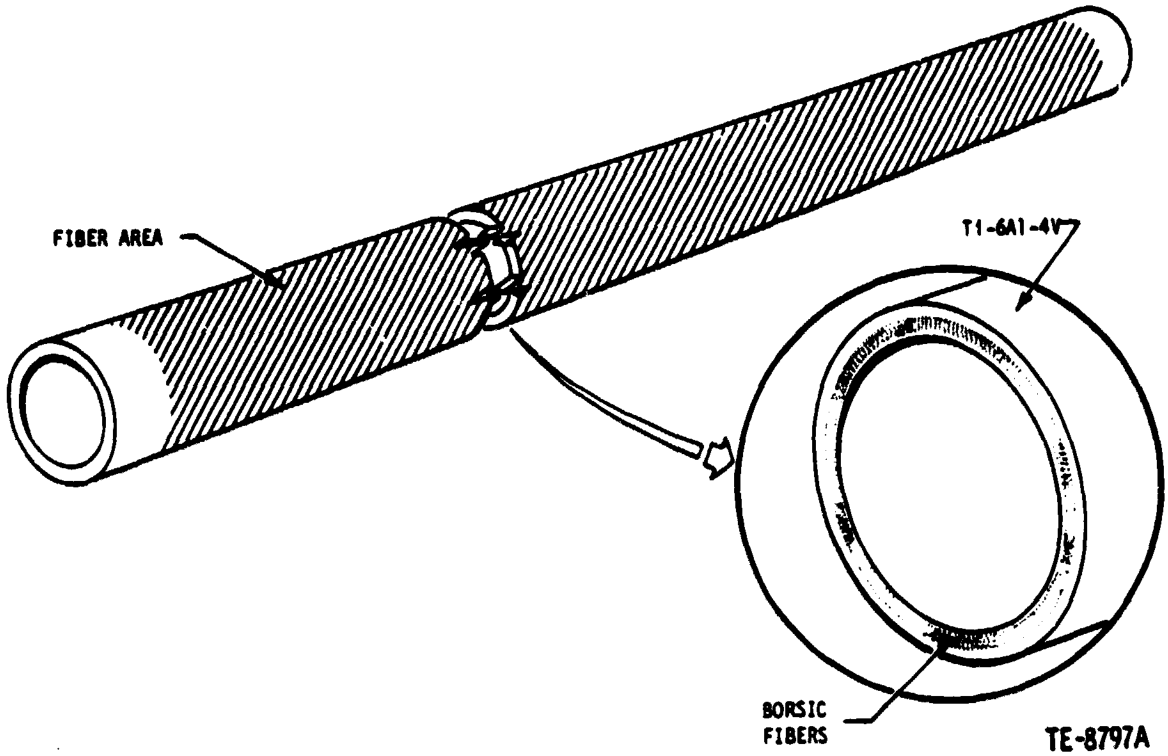

Figure 84. - Typical composite shaft.

This STAT technology research program provides a finite element analysis program capable of computing the engineering properties (E, ρ, σ, τ) for laminates representative of large, metal matrix composite shafts. Sample shafts of this material will be fabricated and tested to verify analytical predictions made with this program. This verified methodology will be an essential tool for the design of advanced STAT engines using composite shafts.

Supercritical Shafting

If the composite shafts are not developed sufficiently by the time they are needed for the STAT engine program, a satisfactory alternate may be available in the form of supercritical steel shafts. These shafts operate above bending critical speeds, as shown in Figure 85. As can be seen, the shafting is configured to operate above two bending critical speeds. Passage through the bending criticals in the transient range is attained through the use of squeeze film dampers located at non-mode locations.

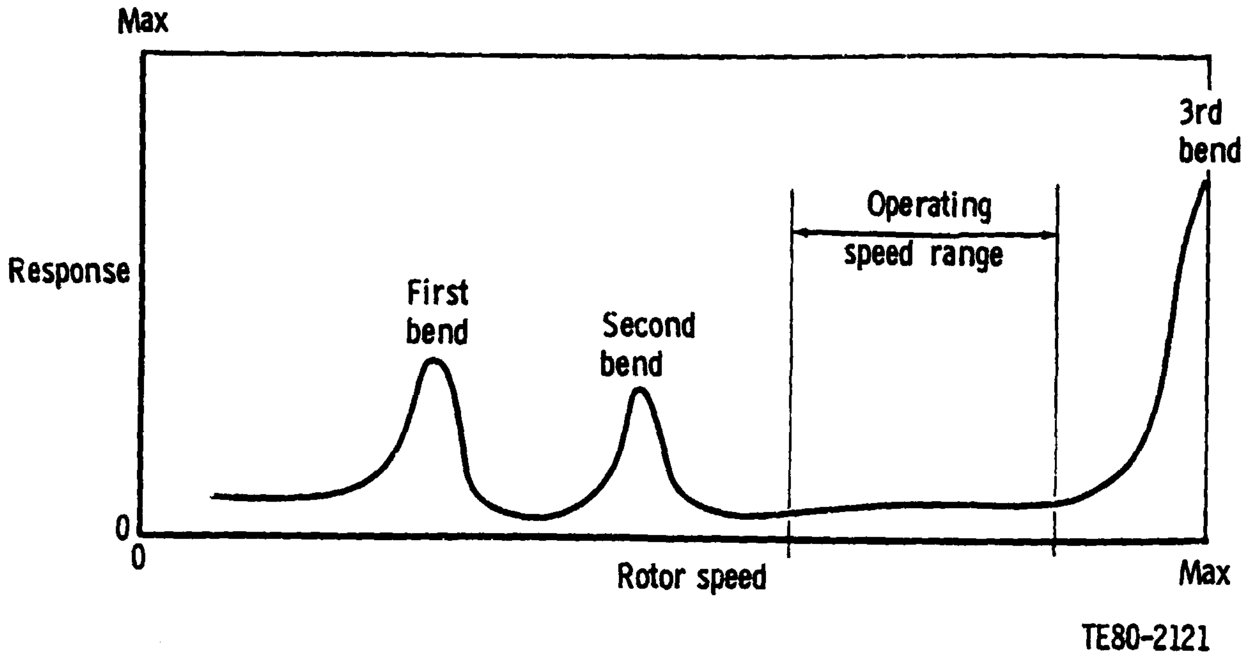

Figure 85. - Typical application of supercritical design to a power turbine shaft.

This STAT technology program will use a preliminary STAT engine design and perform rotor dynamics analyses. Various methods for vibration control of the supercritical shaft will be identified. A rig will be fabricated and tested to evaluate the design and verify the analytical methodology. The results of this program would provide a technique for the application of supercritical shafting to advanced STAT engines if composite shafting is not available.

Bearing Fatigue Life

The high rotor speed and small diameter, which is characteristic of advanced technology, high pressure ratio gas turbines such as the STAT engines, present a problem in designing main bearings with sufficient fatigue life and load-carrying capability. Significant improvements must be made in ball bearings over that which is available for today's gas turbine engines.

The primary contact fatigue failure site on bearing balls has been correlated with end grain concentrations in the polar and equatorial areas, as shown in Figure 86. Reports indicate that ball life also varies with end grain area. End grain varies with grain flow angle relative to the surface. Also, grain areas may well change in size with different ball forging practices and die designs. The equatorial band width may vary with die design and with grind stock allowance.

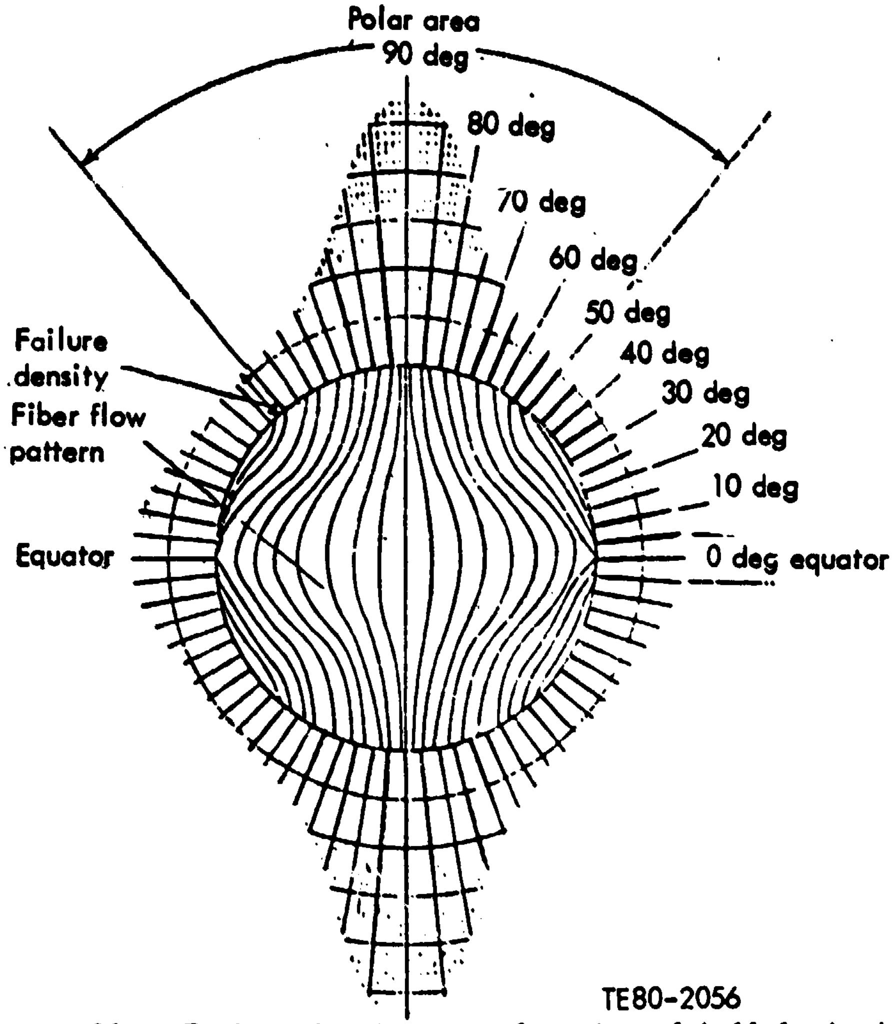

Figure 86. - Failure density as a function of ball latitude.

Thus, standard practices in the forging process and acceptance criteria are needed to obtain maximum ball bearing fatigue life. Fatigue testing of balls with varying end grain characteristics must be performed to achieve fatigue life improvements.

The objectives of this STAT research program are to develop and validate methods of characterizing forging flow lines in bearing components, and to substantiate improvements in fatigue life and load-carrying capability. An assessment would be made of current forging practices for balls and races. Three-dimensional characterization of parts forged by various methods will be performed. Improved forged bearings will be fabricated, and both static tests and fatigue tests will be conducted to quantify the fatigue life improvement.

Diffusers and Combustors

The STAT study showed that one of the biggest factors in reducing commuter transport DOC was the reduction of engine fuel consumption. The advanced VCD, used in the STAT 3579 kW (4800 shp) engine, offers a significant reduction in pressure drop and at the same time a reduction in engine length and weight.

The STAT engines incorporate combustors fabricated from Lamilloy, a DDA-developed, transpiration-cooled structural material. The use of Lamilloy combustors has proven benefits in the areas of cooling air reduction and combustor outlet temperature pattern profile. There is a need to reduce the cost of fabrication of Lamilloy material, as well as to simplify combustor construction techniques, for this material to achieve its full potential for commercial engine production.

The state-of-the-art advances being made in propellers and propfans will result in much lower propulsor noise generation in the STAT transports. It is essential that the STAT engine core noise signature be reduced below that of current technology engines in order to stay below that of the propulsors.

The diffuser and combustor research programs recommended are:

- o Vortex-controlled diffuser
- o Transpiration-cooled combustor

Vortex Controlled Diffuser

The STAT engines must achieve significantly lower fuel consumption than current technology engines. A VCD offers an appreciable improvement in this area but requires further development to ensure the potential gain.

This STAT research effort is aimed at providing needed technical information to assist in the application of the VCD technology to diffuser-combustor systems of advanced STAT engines. The VCD concept features a trapped standing vortex to achieve low pressure loss diffusion in a short length. In order to apply the advanced technology of the VCD concept to future engine designs, it is necessary to investigate and identify the geometric parameters important to system performance. Because of the separated flow nature of the VCD system, analytical modeling of the diffusion process requires elliptic Navier-Stokes numerical schemes, which are in the development stage and require experimental data for verification and improvement.

The proposed research effort will consist of an empirical program to obtain critical design information concerning the influence of selected geometric parameters upon VCD system performance. Results from the test program will be compared to analytical model results, and where discrepancies exist, the model will be improved. the payoffs from this effort will include improved diffuser system performance, reduced diffuser system length, reduced development and manufacturing costs, and improved analytical tools for future design of the STAT engines.

Transpiration-Cooled Combustor

One factor in achieving a low acquisition cost for the STAT engines, and in obtaining longer engine life through a more uniform RIT profile, is in the use of Lamilloy (transpiration-cooled material shown in Figure 87) combustors. The benefits of using Lamilloy for combustor walls have been proved on several Allison gas turbine engines. For this material to be used in commercial engine production, however, it is essential that the cost of fabricating the sheets of Lamilloy material be reduced.

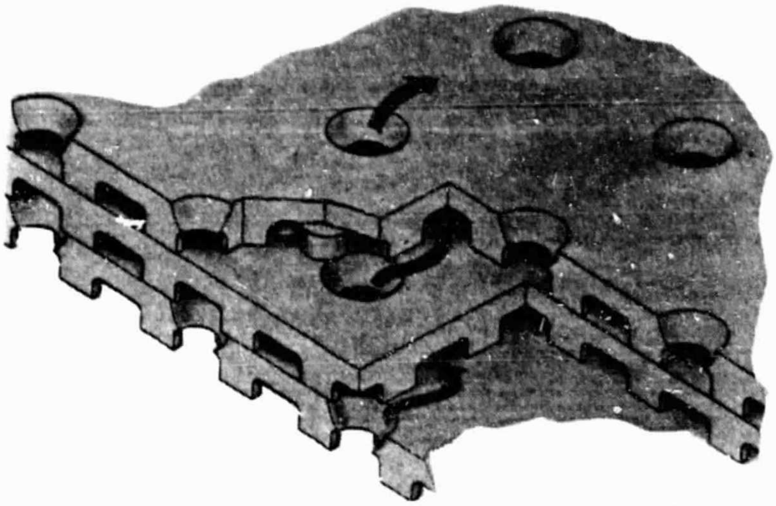

Figure 87. - Lamilloy construction.

This STAT research program would reduce fabrication costs of the Lamilloy material without sacrifice of quality. An expected reduction of 25% in labor is projected with improvements in etching, hole production, sheet cleaning, and bonding procedures. An added improvement would be increased sheet size from 25.4 x 63.5 cm (10 x 25 in.) to 61.0 x 91.5 cm (24 x 36 in.), thus eliminating welds and details in combustor assembly.

Lamilloy combustors are fabricated by forming and joining flat Lamilloy sheets. This additional STAT research program will improve combustor durability, and thereby increase STAT engine life, through assembly improvements. These improvements will be made in the areas of welding (to produce a narrow bead), forming (to reduce flow restriction), and nondestructive testing methods. These fabrication improvements will produce lower stress and temperature values in joints and at radii.

Engine Accessories

The STAT studies gave a clear indication that the two major areas, wherein engine improvements would be most beneficial to reducing commuter aircraft DOC, were in reduction of fuel consumption and maintenance cost. With regard to lowering turboprop maintenance costs, an area with great potential is in the application of an all-electronic fuel control system integrated with an engine condition monitoring system permitting full on-condition maintenance. Three areas requiring state-of-the-art advances for the practical application of such a system to STAT engines are:

- o Electronic control for advanced, variable geometry, turboprop engine
- o Fuel pumping and metering system
- o System integration of electronic control, fuel pump and metering system, condition monitoring system, and propeller/propfan control

The engine accessories research programs recommended are:

- o Electronic control
- o Fuel pumping and metering system
- o Engine condition monitoring

Electronic Control

To meet the STAT program goal of a significant reduction in maintenance cost over current technology engines, an advanced, full-authority, digital electronic fuel control system is a must. Such an integrated control system, as shown in Figure 88, would provide for constant optimum operation of the engine and propeller plus provide condition monitoring of both.

The proposed STAT research program is the first step in a total control development program. It is structured to address the advance turboprop requirements associated with failure modes, manual control, and total system integration, including the engine condition monitoring system.

Fuel Pump and Metering System

An all-systems integrated, electronic control system is mandatory on STAT engines to achieve the low maintenance cost and high reliability desired. The objective of this STAT technology research effort is the design and partial development of an advanced fuel pump and metering system specifically suited for use in a full-authority digital electronic control system.

In order to realize maximum benefits in performance improvements, while reducing overall system cost, reducing weight, and improving reliability, the fuel handling portions of the system must have design features and characteristics fully compatible with the overall system requirements.

DDA is currently evaluating five new fuel pump and metering system concepts specifically configured for use in future applications such as the STAT engines. This current program, funded under Contract NAS3-22046, has defined the requirements for such a fuel-handling system and developed possible approaches for tradeoff studies. These studies define the following factors for making a comparative assessment of the candidate systems:

- o Reliability
- o Cost
- o Weight and size
- o Maintainability
- o Performance
- o Back-up operation
- o Development risk

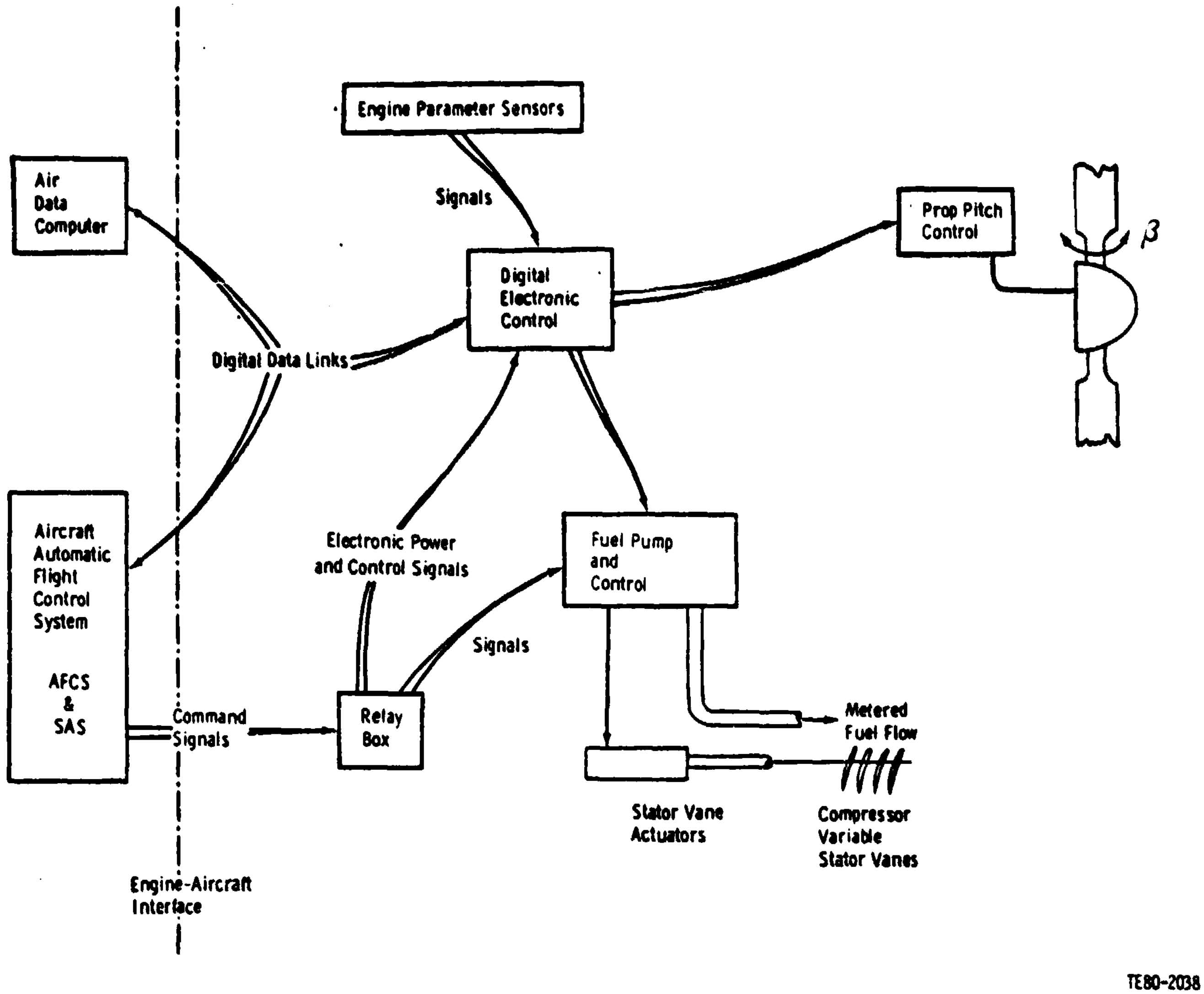

Figure 88. - Advanced turboprop propulsion control system.

From these assessments, a system is to be selected and a preliminary design developed. The recommended STAT research program would be a direct follow-on to this currently funded work to proceed with hardware development and proof tests.

This STAT program would include the detailed design of an advanced fuel pump and metering system. Fabrication of hardware and bench testing would be performed to determine performance capabilities. After bench evaluation, the system would be evaluated on a Model 250 gas turbine engine during test stand operation, and would be tested in conjunction with an existing full-authority digital electronic controller.

Flight testing of the total system model on a 250 engine would be performed to evaluate the dynamic performance under the actual aircraft operating environment.

Engine Condition Monitoring

The STAT commuter aircraft requirements for propulsion, high reliability, and low maintenance cost would be met in part by the complete integration of an engine condition monitoring system with the engines' basic electronic fuel and prop control systems.

This STAT engine condition monitoring program will develop improved methodologies in order to support total on-condition maintenance concepts. An engine condition monitoring system, such as that shown in Figure 89, will continuously monitor all engine and propeller operation to determine engine life usage, and to provide maintenance directives and safety warnings. Life usage based on actual operation permits vastly improved usable life within acceptable risk limits. Automatic maintenance directives result in timely repairs, while reducing unrequired maintenance activity. Accurate safety warnings lower the risk of aircraft operation.

In particular, this program will develop and demonstrate improved techniques relative to:

- Mechanical LCF accounting
- Turbine thermal stress analysis
- Performance degradation analysis

LCF and turbine thermal stress analysis enable part-life prediction to be based on actual usage rather than worst-case time estimates. This feature permits safely running parts nearer their theoretical life, and therefore greatly improving service life. Performance degradation analysis complements the true on-condition maintenance philosophy.

The techniques resulting from this STAT research program will enable incorporation of an on-condition monitoring system for the next generation STAT engines.

Noise Reduction

Compressor Noise Reduction

High cycle pressure ratio advanced gas turbine engines use high speed compressors, such as shown in Figure 90, with supersonic blade tip velocities. The resulting multiple pure-tone (MPT) generation would add significantly to the noise signature of the STAT commuter aircraft equipped with a propfan or advanced, low noise propeller. This effect is expected to be particularly acute during approach, when propeller thrust is low.

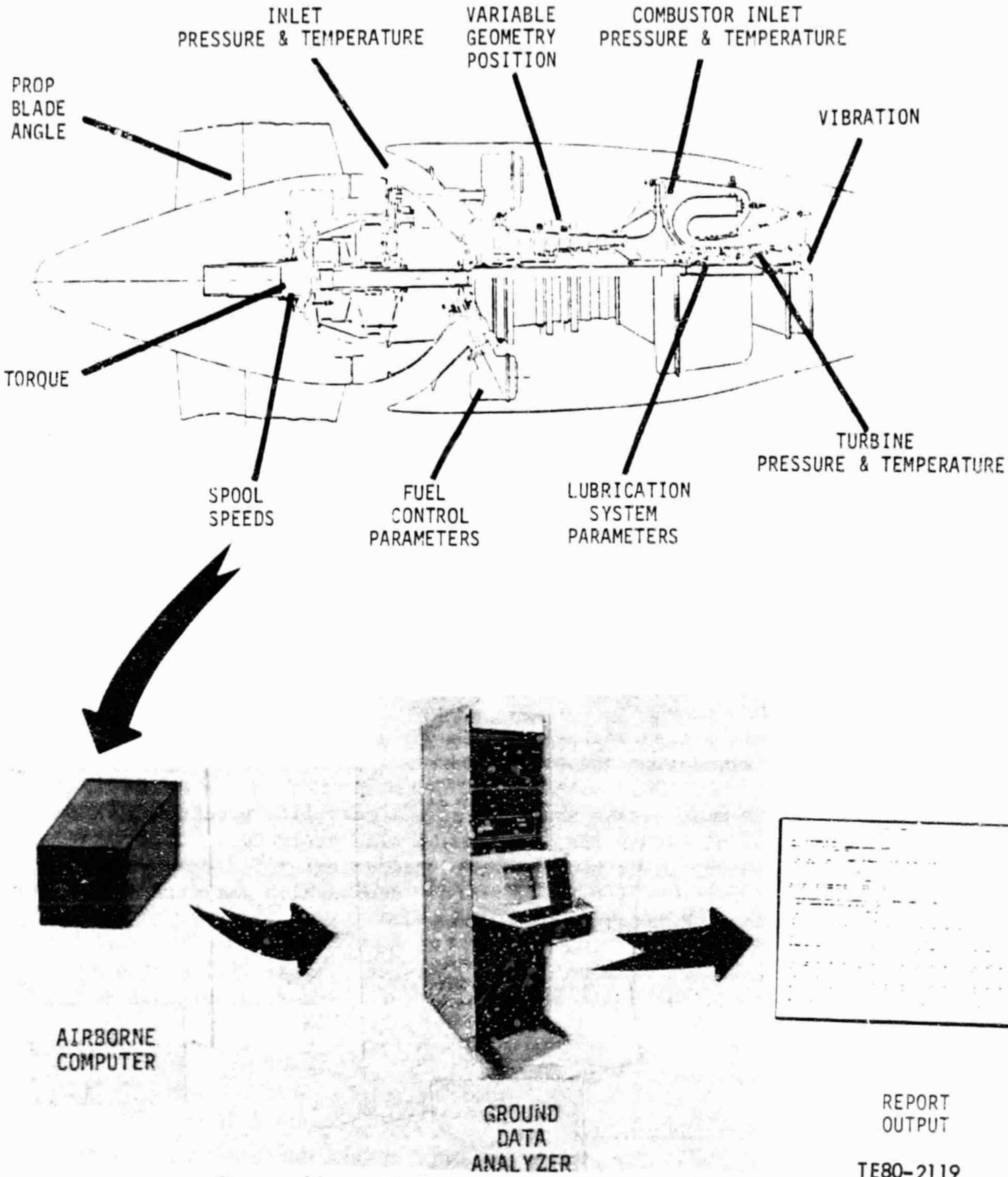

Figure 89. - STAT engine condition monitoring.

This proposed STAT research program will validate the theory that leading edge sweep materially reduces the effective Mach number at that location, and thereby reduces the noise generated. This theory can be proven by test of either an axial or centrifugal compressor. An existing single-stage centrifugal compressor rig will be used for this research program. The inducer will be redesigned to provide leading edge sweep and used to fabricate a test unit. A test will be conducted of both original and modified configurations.

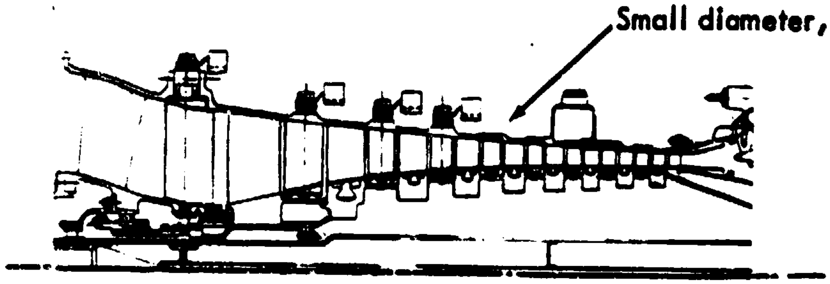

Figure 90. - Typical advanced high pressure ratio compressor.

The results of this test should verify the theory in question and provide a data base for designing advanced gas turbine compressors without MPT noise generation.

Combustion Noise Reduction

Combustion-generated noise (see Figure 91) usually referred to as "core" noise, has been a limiting factor in achieving large turbofan noise reductions in the low thrust approach condition where jet and fan noise are lowest. For many turboshaft engines, the combustor is the dominant noise generator, and combustion noise represents the major portion of the total sound power radiated at both high and low power settings. Engine noise has generally been assigned a minor role in assessing noise generated by turboshaft propulsion systems because of the obvious high level harmonic noise generated by propellers and rotors, and the difficulty in separating engine/propeller/rotor broad band noise. Studies at DDA indicate that the contribution of engine combustion noise is not minor, and in fact, may be a major obstacle to designing new turboprop transport aircraft to meet certification noise requirements. This will be particularly true for the STAT transports where advanced propulsors are being developed with very low noise signatures.

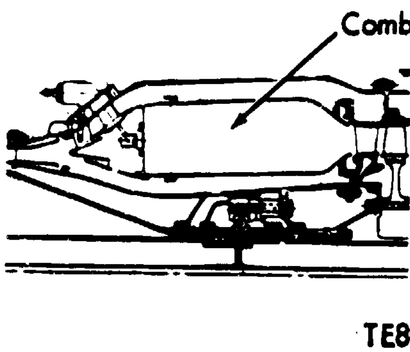

Figure 91. - Typical gas turbine annular combustor.

The recommended STAT research program addresses the problem of low noise combustor development. A Model 250 engine will be tested to determine its noise signature in combination with select performance characteristics. An analysis of this data would be made, and combustor design modifications formulated for reduced noise generation. These design modifications would be tested on a Model 250 combustor rig to obtain correlating experimental data. The optimum combustor configuration would be tested on a Model 250 engine to verify the reduced noise signature.

The information gained from this STAT research program would enhance the ability to design advanced gas turbine engines with lower noise combustors.

Reduction Gears

The metal matrix composite materials now available for the reduction gear and other major engine castings, offer significant gains in weight reduction while providing increased thermal stability and greater rigidity. Before these materials can be used advantageously in the STAT engines, however, much information is required relative to their material properties and machinability. The reduction gear research program recommended concerns the use of composite gear cases.

Composite Gear Cases

One way to reduce the DOC of STAT engines is to lower their weight, compared with current technology engines. A considerable weight reduction potential exists in the use of new metal matrix composite materials for the reduction gear cases as shown in Figure 92. Composite gear cases will also provide greater rigidity and increased thermal stability for the reduction gear assemblies.

In this STAT research program, DDA would investigate two metal matrix composite materials for this application: SiC whiskers in magnesium or aluminum castings, and chopped polycrystalline alumina (FP) fibers in magnesium or aluminum castings.

In order to use this material in the design of the STAT engines, material characterization of each composite must be performed. Mechanical properties must be determined, including stiffness, strength, thermal coefficient, LCF, HCF, and corrosion resistance. Also, the machinability of the new materials must be determined in order to successfully fabricate the advanced engine gearboxes. Machining characteristics such as turning, boring, threading, etc., will be evaluated using test specimens.

EXPERIMENTAL ENGINE PROGRAM

Full-scale development of a STAT engine is viewed as a commercial risk venture dictated by market requirements. Technical and economic decisions are required to establish engine size. Such items as passenger load, unrefueled range with reserves, flight speed, cruise altitude, and field length strongly impact engine size. The R&D programs recommended will provide design data to permit selecting the proper components and arrangement for the size ultimately

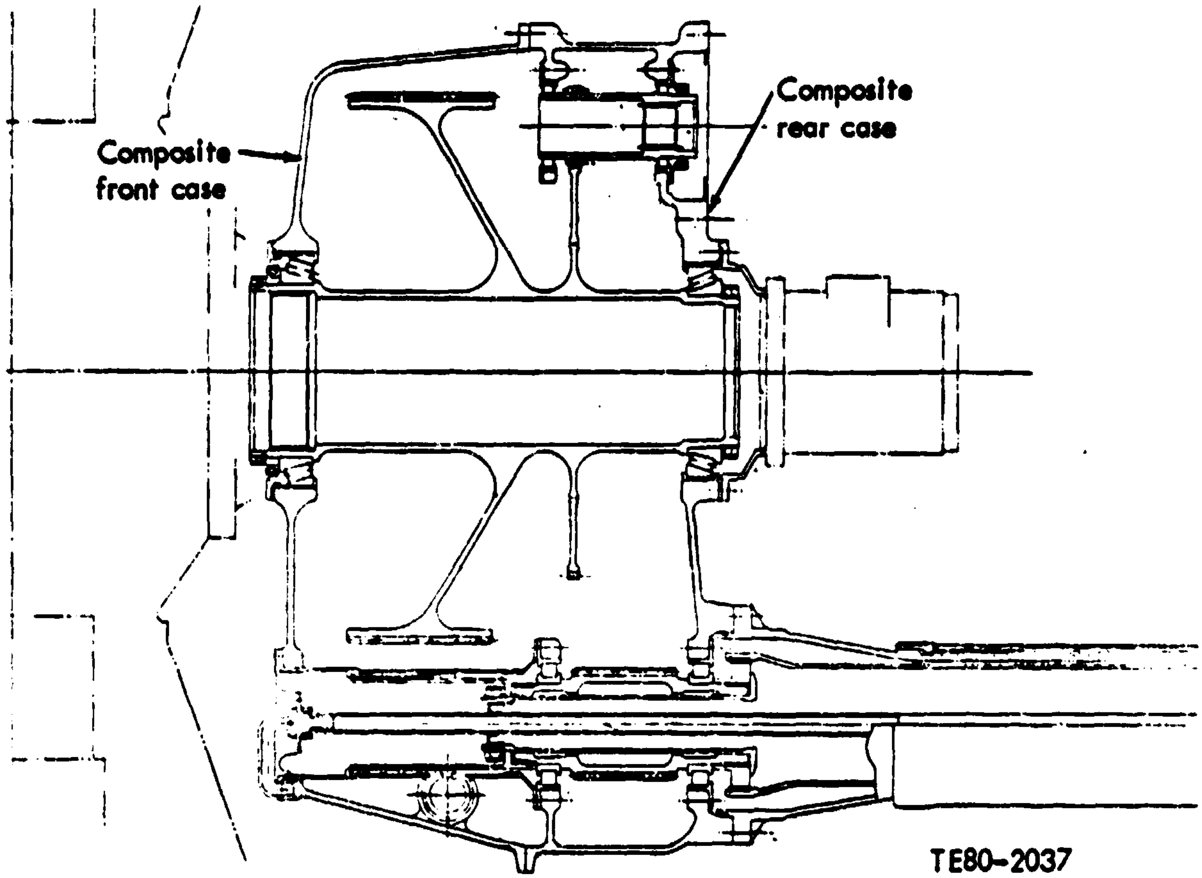

Figure 92. - 3579 kW (4800 shp) STAT engine reduction gear.

required. Technology needs can sometimes be developed in larger sizes; however, it is generally recognized that risk is reduced appreciably if suitably constructed experiments can be accomplished in the size and environment of an appropriately sized experimental engine.

Such an experimental engine program would also offer an unparalleled opportunity to test unique technology features and systems being newly incorporated in the STAT engines. An example of this would be the active turbine clearance control system described in the subsection entitled, "Configuration Trades." In this instance, the technology for such a system exists now, but the mechanical integration is best accomplished on an experimental engine prior to full commercial development.

Table LXX lists the 21 STAT technology research programs recommended to NASA as a result of this study effort. This table indicates 10 research programs which would benefit from the experimental engine program, since the applicable technologies are best developed on the unity size engine.

Upon completion of basic R&D programs of the type described in the previous section of this report, it will then be feasible to conduct an experimental STAT engine program. The results of this program will provide assurance that full commercial development of the STAT engine may be initiated within an acceptable risk.

TABLE LXX. - RECOMMENDED STAT TECHNOLOGY PROGRAMS

Technology research item	Nonscalable	Scalable	Comment
Compressors			
Hybrid centrifugal compressor impeller	X		Centrifugal impeller used only in small engines
Compressor erosion protection		X	
Inlet particle separators	X		Engine size related to problem
Compressor noise reduction	X		Size and high speed related to problem
Axial compressor aft stage study	X		Aft stage blading size critical
Centrifugal compressor-advanced high hub/tip	X		Centrifugal impeller used only in small engines
Rotor/case response to rotating stall		X	
Turbines			
Hybrid rotors	X		Used in small engines only and size is critical
Cast-in impingement cooling	X		Problem/solution occurs in small size only
Abradable coatings	X		Minimal clearances relate to small blades
Diffusers-combustors			
Vortex-controlled diffuser		X	
Transpiration-cooled combustor-sheet fab.		X	
Transpiration-cooled combustor-fab.		X	
Combustor noise reduction		X	
Structure and shafting			
Composite gear cases		X	
Composite shafting	X		Large engine shafts use different composite
Bearing fatigue life		X	
Supercritical shafting	X		Applicable to high speed only, dampers size critical
Engine systems			
Electronic control		X	
Fuel pump and metering system		X	
Engine condition monitoring		X	

The basic elements and timing of such an experimental engine program are shown in Figure 93. After six months of design effort, a "long lead time" release of selected engine hardware will be made. The full detailed release of all hardware will be achieved nine months from go-ahead.

Three serialized test engines would be built for this program, with the first delivered in 22 months. Three equivalent engine's worth of spare parts would be fabricated to support the component and engine test programs.

The component development required for this experimental engine program would entail three basic types of rig operation. A full-scale combustor rig program would be initiated 10 months from go-ahead. The combustor testing would require approximately 50 hours of rig time, and would cover cold flow aerodynamic and pressure drop tests and burning tests of the following types:

- o Thermal paint (metal temperatures)
- o Sea level and altitude relight characteristics
- o Fuel distribution
- o Power calibration

Approximately 100 hours of compressor rig tests would be conducted to determine aerodynamic performance in terms of airflow, pressure ratio, and efficiency, as well as interstage data to define the operating characteristics of each individual stage.

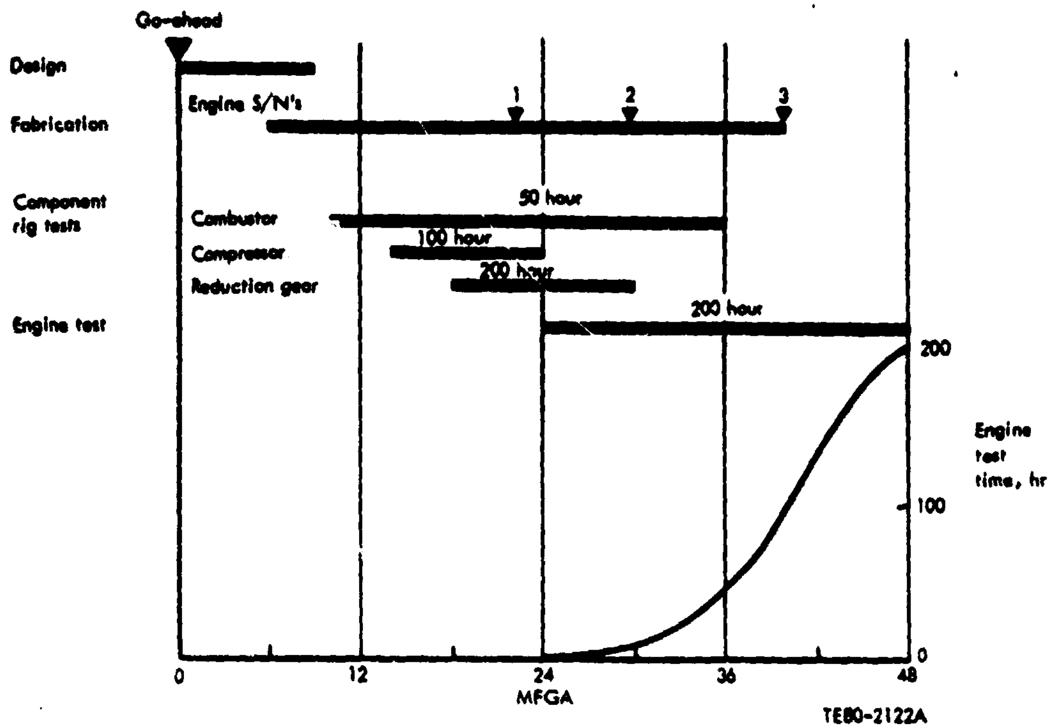

Figure 93. - STAT experimental engine schedule.

The reduction gear assembly would be tested on a back-to-back rig, wh ch is capable of simulating full-design torque at rated speed. Approximately 200 hours of testing will be conducted to determine gear profile characteristics, lubrication capability, and structural stability of the gearbox.

The engine testing consists of three types: gas producer, power section, and full engine. The gas producer tests would provide the initial performance verification of the HP turbine. The power turbine performance would be determined from the power section tests. The power section tests would be conducted on a dynamometer to provide full power absorption and checkout of all mechanical and lube systems. The engine testing would be conducted on a prop stand to provide for complete fuel and control system checkout. A total of 200 hours would be accumulated during these tests.

This program, with a parallel prop fan program and aircraft studies, will provide the basis for launching an integrated demonstration effort, which would then be followed by a commercial aircraft system development program.

CONCLUSIONS

STAT studies show that the largest improvement in commuter aircraft DOC, as influenced by engine characteristics, is achieved through reduced fuel consumption and reduced maintenance cost. A 10% reduction in sfc results in 3.5 to 4.0% reduction in DOC for the aircraft used in these studies when fuel is priced at $0.264/L ($1.00/gal). A 10% reduction in maintenance cost was found to be worth 1.0 to 1.3% in DOC.

Fuel consumption can be reduced through improved thermodynamic cycle efficiency. A compresor pressure ratio of approximately 20:1, and an RIT near 1506 K (2250°F), appears to be best for minimum DOC.

Compressors chosen for STAT engines feature single-spool configurations with a reduced number of stages and reduced blade count, compared to current technology engines. Compressor-related research programs recommended for technology development include:

- o Hybrid centrifugal compressor impeller
- o Compressor erosion protection
- o Inlet particle separators
- o Compressor noise reduction
- o Axial compressor aft stage study
- o Centrifugal compressor advanced high hub/tip ratio study

Hybrid turbine wheels are needed with greatly increased bore load capability. Composite shafting with increased stiffness is required to permit subcritical operation with simple-rotor support. Dynamic problems associated with rotor/case response to compressor rotating stall must be understood to optimize compressor vane and blade tip clearance and ensure structural rigidity. Interstage and blade tip seal development must be continued if desired turbine efficiency levels are to be attained. Turbine, bearing, shafting, and dynamics research programs recommended for technology development include the following:

- o Hybrid rotors
- o Cast-in impingement cooling
- o Abradable coatings
- o Composite shafting
- o Bearing fatigue life
- o Rotor/case response to rotating stall

Fuel system maintenance is a major contributor to problems in achieving consistent engine availability, and is also a significant cost factor. The electronic control system concept offers the potential of solid improvement in reliability and maintainability, as well as interfacing with an on-condition maintenance data system to predict timely maintenance actions.

Maintenance costs can be reduced by incorporating on-condition maintenance, improved modularity, improved reliability and maintainability, and an efficient maintenance management system using engine condition-monitoring data for key input.

R&D programs recommended in this area include:

- o Electronic control
- o Fuel pump and metering system
- o Engine condition monitoring

Incorporation of the technology advances recommended for the STAT engine will result in the following improvement in engine characteristics referred to the current technology base used in the study:

Power size, kW (shp)	1790 (2400)	3579 (4800)
sfc--%	-16.8	-19.0
Weight--%	-12.6	-25.3
Price--%	-18.9	-16.4
Maintenance cost--%	-56.1	-62.0
Envelope length--%	-21.8	-16.7

Achieving these improvements will result in significant fuel savings and reduction in DOC compared with the current technology base, as shown below:

Aircraft	Low speed	High speed
Fuel consumption--%	-19.6	-22.8
DOC*--%	-13.6	-16.2
Total cost of ownership*--%	-10.9	-13.3

*$0.396/L ($1.50/gal)

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3. Parametric Propeller Data Package for Advanced Technology Commuter Aircraft Propellers, Small Transport Aircraft Technology Propeller Study, prepared for NASA-Lewis by Hamilton Standard Div., United Technologies Corp., under Contract NAS3-22039.

LIST OF ABREVIATIONS AND SYMBOLS

Symbol	Meaning
AFSC	Airframe specific cost
A	Axial
A-C	Axial-centrifugal
ATE	Advanced technology engine
AEO	All engines operative
AR	Aspect ratio
A_w	Area wetted (aircraft)
ATA	Air Transport Association
ALT	Altitude
AN^2	Blade stress parameter
AFWT	Airframe weights
AF	Adjustment factor
ATEGG	Advance turbine engine gas generator
BOT	Burner outlet temperature
Burner Δp	Combustion pressure drop
BSFC	Brake specific fuel consumption
CPR	Compressor pressure ratio
'CTE'	Current technology engine (without inherited learning)
CTE	Current technology engine (with inherited learning)
CF	Conversion factor
C_D	Coefficient of drag
Cent	Centrifugal
DOC	Direct operating cost
DDA	Detroit Diesel Allison
dB	Decibel
Dp	Diameter (propeller)
DTE	Derivative technology engine
EAS	Equivalent airspeed
EFH	Engine flight hour
E	Young's modulus
ESFC	Equivalent specific fuel consumption
EPNL	Effective perceived noise level
E/ρ	E/rho (Young's modulus-to-density ratio)
Ep	Engine power
EPNdB	Effective perceived noise-decibels
EMDP	Engine model derivative program
F_{NET}	Net jet thrust
F_{TOT}	Total engine thrust
FOD	Foreign object damage
FAR	Federal Aviation Regulation
FW	Finished weight
F	Fahrenheit
FC	Fixed costs

Symbol	Meaning
gal	Gallon
GE	General Electric
GBL	Gearbox loss
GB	Gearbox
$gJ\Delta h/U^2_{mean}$	Turbine average stage loading
HIP	Hot isostatic press
HLH	Heavy lift helicopter
HP	High pressure
HPT	High pressure turbine
HCF	High cycle fatigue
h	Hour(s)
IOC	Initial operation capability
IFR	Instrument flight rules
IAS	Indicated air speed
Imp	Impingement
I_c	Index of cost
ID	Inside diameter
IPS	Inertial particle separator
KIAS	Knots indicated air speed
K	Kelvin
L	Litre
LCC	Lockheed California Company
L/D	Lift/drag (ratio)
LCF	Low cycle fatigue
LP	Low pressure
LPT	Low pressure turbine
LD	Blade-to-blade shroud loading
M_N	Mach number
MPA	Maritime partrol aircraft
Mfg	Manufacturing
MIF	Materials index factor
MTBF	Meantime between failures
MTBR	Meantime between removals
MPT	Multiple pure tone
NA	Not available
NTS	Negative torque signal
NPR	Nozzle pressure ratio
$N/\sqrt{\theta}$	Corrected rotational speed
N_{PT}	Power turbine rotational speed
NGGT	Gas generator turbine rotational speed
N	Rotational speed
N_S	Specific speed
OASPL	Overall sound pressure level
OEW	Operating empty weight
OEM	Original equipment manufacturer
OEI	One engine inoperative
OD	Outside diameter

PC	procurement costs
PPFRT	Prototype preliminary flight rating test
PAX	Passenger
PS	Power section
P&W	Pratt and Whitney
PRR	Premature removal rate
P_x	Power extracted
RIT	Rotor inlet temperature (turbine)
R_{comp}	Compressor pressure ratio
R_c	Compression ratio
R_b	Radius of bore (wheel)
R_o	Radius of outer diameter (shaft)
R&D	Research and development
sfc-un	Specific fuel consumption (uninstalled)
sfc	Specific fuel consumption
SLSS	Sea level static, standard day
SF	Scale factor
SLS	Sea level static
SLTO	Sea level takeoff
shp	Shaft horsepower
shp-un	Shaft horsepower (uninstalled)
snm	Seat nautical mile
skm	Seat kilometer
TOGW	Takeoff gross weight
TCO	Total cost of ownership
TAC	Total aircraft cost (flyaway price)
Ti	Titanium
T_c	Temperature of cooling air
TBO	Time between overhaul
TO	Takeoff
T_{sfc}	Thrust specific fuel consumption
U	Utilization (rate)
U_t	Tip tangentrial velocity
$U_t/\sqrt{\theta}$	Corrected tip speed
VB	Block velocity
VCD	Vortex controlled diffuser
V/STOL	Vertical/short takeoff and landing
$W\sqrt{\theta}\,\delta$	Corrected airflow
W_c	Cooling airflow
W_a	Airflow
W_f	Fuelflow

Greek Alphabet

η	Eta (efficiency)
η_p	Propeller efficiency
σ	Sigma (tensil strength)
τ	Tau (shear strength)
$\Delta P/P_{burn}$	Combustor pressure drop
$\Delta h/\theta_{cr}$	Stage equivalent work
η T-T	Total-to-total adiabatic efficiency
ρ	Rho (weight density)
σ/ρ	Tensile stress-to-density ratio
τ/ρ	Sheer stress-to-density ratio