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STUDY OF ADVANCED PROPULSION SYSTEMS FOR SMALL TRANSPORT AIRCRAFT TECHNOLOGY (STAT)

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A Division of The Garrett Corporation

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16. Abstract

The study was conducted to identify technology requirements and define the research and development required for new commuter aircraft turbine engines. Definition of takeoff gross weight, performance, and direct operating cost for both a 30- and 50-passenger airplane was established. The study indicated that a potential direct operating cost benefit, resulting from advanced technologies, of approximately 20 percent would be achieved for the 1990 engines. Of the numerous design features that were evaluated, only maintenance-related items contributed to a significant decrease in direct operating cost. Recommendations are made to continue research and technology programs for advanced component and engine development.

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1.0 SUMMARY

This NASA Small Transport Aircraft Technology (STAT) Propulsion Study was conducted to identify technology requirements and define the research and development required for new commuter aircraft. The study was divided into three major tasks:

- Definition of baseline aircraft and propulsion systems
- Identification and assessment of potentially beneficial advanced propulsion technologies and design features
- Recommendation of future research to develop and implement the beneficial technologies and features.

Definition of the takeoff gross weight (TOGW), performance, and direct operating cost (DOC) for both a 30- and 50-passenger airplane was established as follows for 1980 technology levels:

	30-Passenger Airplane	50-Passenger Airplane
TOGW	13,747 kg (30,308 lbm)	19,434 kg (42,845 lbm)
SHP/Engine	1373.5 kW (1842 hp)	1811.3 kW (2429 hp)
DOC for 100-nmi Mission	$0.097/seat nmi	$0.078/seat nmi

Baseline engines representing current production technology (e.g., TPE331-11) were defined as shown in Table I. Comparable engine characteristics for 1985 technology derivative engines and 1990 technology engines are also shown in Table I. It was determined during the study that the potential benefit resulting from advanced 1990 technologies is a reduction in DOC of approximately 20 percent.

The large potential DOC improvements identified in this study warrants initiation of NASA-sponsored research and technology programs for appropriate components. This includes such items as compressors, combustors, turbines, gas generators, and demonstrator engines. NASA sponsorship of the integrated development of these components with a demonstration in an experimental engine program would provide the impetus for industry to undertake the development and production of the STAT engines.

TABLE I. STAT ENGINE CHARACTERISTICS[1] FOR DIFFERENT TECHNOLOGY LEVELS, FIXED MISSION, VARIABLE AIRCRAFT.

PARAMETER	SI UNIT	CUSTOMARY UNIT	1980 BASELINE				1985 DERIVATIVE				1990 STAT TECHNOLOGY			
			30-PAX A/C		50-PAX A/C		30-PAX A/C		50-PAX A/C		30-PAX A/C		50-PAX A/C	
			SI VALUE	CUST. VALUE	SI VALUE	CUST. VALUE	SI VALUE	CUST. VALUE	SI VALUE	CUST. VALUE	SI VALUE	CUST. VALUE	SI VALUE	CUST. VALUE
Shaft horsepower	kW	hp	1373.5	1842	1811.3	2429	1359.4	1823	1759.6	2360	1343.8	1802	1777.7	2384
BSFC	kg/kW·h	(lbm/hr)/hp	0.3376	0.558	0.3376	0.555	0.3054	0.502	0.3054	0.502	0.2683	0.441	0.2683	0.441
Weight[2]	kg	lbm	345.2	761	520.3	1147	295.7	652	390.5	861	282.6	623	373.8	824
1980 OEM Cost/engine	$ x 1000	$ x 1000	230.0	230.0	304.0	304.0	242.5	242.5	320.3	320.3	241.0	241.0	318.1	318.1
SHP/WT	kW/kg	hp/lbm	3.978	2.42	3.485	2.12	4.603	2.80	4.603	2.80	4.751	2.89	4.751	2.89
SHP/WA	kW/(kg/s)	hp/(lbm/sec)	214.87	130.7	214.87	130.7	237.39	144.4	237.39	144.4	335.87	204.3	335.87	204.3
Cost/SHP	$/kW	$/hp	167.49	124.9	167.90	125.2	178.36	133.0	178.36	133.0	179.29	133.7	178.89	133.4

[1] Sea level static, standard day, takeoff power, uninstalled.

[2] Includes gearbox, controls and accessories.

2.0 INTRODUCTION

Commuter airlines are a rapidly growing segment of the American aviation industry and increased opportunities for growth have been afforded under the provisions of the Airline Deregulation Act of 1978. This growth may require new, higher-capacity aircraft with superior aerodynamics, modern structures, better flight-control systems; and substantially-improved, fuel-efficient, low-maintenance propulsion systems.

At $0.264/l ($1.00/gal), fuel accounts for approximately 35 percent of the typical commuter DOC, and will account for more than one-half of the DOC if fuel costs reach $0.528/l ($2.00/gal). Engine maintenance costs account for approximately 10 percent of the DOC. NASA, in recognition of the importance of the propulsion system and its effect on DOC, sponsored this STAT study with the general objective of identifying advanced propulsion system technologies that could improve the performance and economics of current-technology commuter aircraft. The study was directed toward hypothetical, all-new, 1990-time-frame engines, with comparisons against current-technology and mid-1980's derivative engines.

The STAT study was a 16-month effort divided into three major tasks:

- Definition of baseline aircraft and propulsion systems
- Identification of potentially benefical advanced propulsion technologies and design features, and the assessment of their benefits with respect to performance, weight, and cost
- Recommendation of future research to develop and demonstrate the technologies and features that were identified as having significant potential.

3.0 DEFINITION OF BASELINES

Baseline commuter aircraft and missions were defined for the study by the NASA-Ames Research Center to provide a benchmark for evaluating advanced propulsion technology. These aircraft represented existing airframe and propulsion-system technology.

3.1 Airplanes and Missions

The 30- and 50-passenger airplanes were designed to the requirements listed in Table II. As indicated, the airplanes were designed for a 600-nmi mission, plus reserves; however, a 100-nmi mission was also defined and was used for economic evaluations performed later in the study. The mission profiles are shown in Figure 1.

The two airplane configurations and their principal characteristics are shown in Figures 2 and 3. The three views in these figures are to the same scale so that the relationship between the two configurations can be seen. Additional details of the airplane designs, in the form of copies of the computer printouts provided by NASA-Ames, are contained in Appendix I.

An important part of the baseline airplane definitions was the derivation of the sensitivity of airplane characteristics such as DOC, TOGW, empty weight (EW), block fuel and aircraft acquisition cost to engine SFC, weight, engine cost, maintenance cost, and reliability. The sensitivities of DOC for the 30- and 50-passenger airplane designs to engine SFC, weight, engine costs, and maintenance costs are shown in Table III. These sensitivities reflect an iterated design. For example, if engine SFC is reduced, the new DOC represents a lower TOGW, EW, horsepower, etc. These sensitivities indicate the strong influence of SFC, as is subsequently presented in the discussion of DOC. Application of these sensitivities shows that a 1-percent change in DOC (approximately) results from:

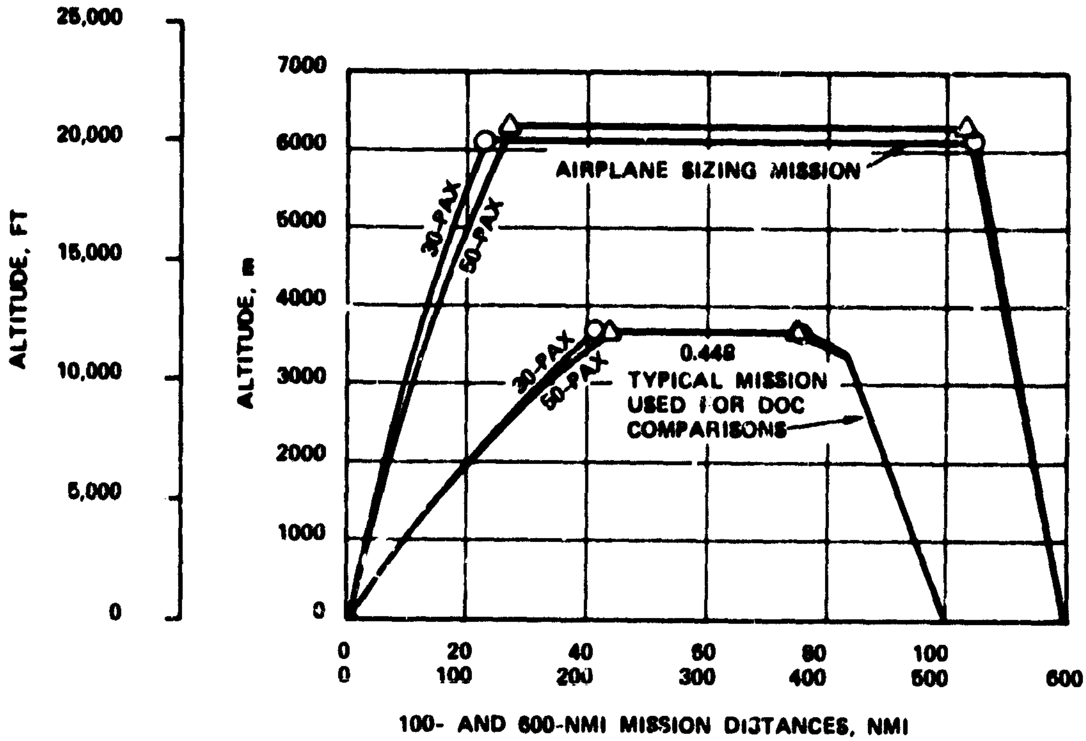
- A 2-percent change in SFC
- A 10-percent change in engine maintenance cost
- An 18-percent change in engine weight
- A 35-percent change in engine initial cost (not including maintenance cost effect).

3.2 Engines and Propulsors

The engines defined for the study were based on technology levels equivalent to the 1980, 1985, and 1990 time periods. The 1980-technology engine is the Garrett Model TPE331-11 turboprop, shown in cross section in Figure 4. The basic performance characteristics of this engine are given in Table IV. The complete

TABLE II. BASELINE AIRPLANE ASSUMPTIONS AND REQUIREMENTS.

- Scaled versions of either existing turboprop engines or representative hypothetical equivalents.
- 90.7 kg (200 lb) per passenger including baggage.
- 2-man crew at 90.7 kg plus 1 flight attendant at 59 kg (130 lb) (per 50 passengers).
- 1.8 m (6 ft) minimum interior aisle height.
- Minimum 81.3-cm (32-inch) seat pitch, 45.7-cm (18-inch) seat width between armrests, and 45.7-cm aisle width.
- 0.14 m^3 (5 ft^3) per passenger for easily loaded preloaded baggage storage, plus carry-on baggage provision of 50.8 cm x 50.8 cm x 27.9 cm (20 in. x 20 in. x 11 in.) per passenger; and garment storage area of 2.0 cm (0.8-inch) width per passenger.
- One lavatory per 50 passengers.
- 34.5 kPa (5 psi) cabin pressurization minimum.
- Maximum cabin interior noise level less than 85 dB OASPL, and speech interference level of less than 65 dB.
- Airframe design life of at least 30,000 hours and 60,000 takeoff and landing cycles.
- Full design payload to be carried over a range of 600 nmi with instrument flight rules (IFR) reserves for a 100-nmi alternate, and 45 minutes at maximum endurance power at 3050-m (10,000-ft) altitude.
- Field length shall not exceed 1220 m (4000 ft) for a hot day [305.4 K (90°F)] at sea level, per FAR 25.
- Aircraft shall meet current Federal Aviation Regulations (FAR) 36 Stage 3 noise limits minus 8 EPNdB at all measurement locations.
- A cruise-speed capability of at least 250 knots indicated airspeed at 1830- through 3050-m (6000- through 10,000-ft) altitudes, standard-day conditions.
- A terminal area speed capability of at least 180 knots indicated airspeed with gear and flaps extended in order to stay with large jet aircraft.
- A stall speed less than 93 knots in landing configurations at maximum landing weight in order to qualify for operations in Instrument Approach Category B aircraft requirements.



FLIGHT SPEEDS

NO. OF PAX	30		50	
RANGE, NM	600	100	600	100
CLIMB, KEAS	225	225	223	223
CRUISE, M	0.461	0.452	0.456	0.448
DESCENT, KEAS	240	233	240	233

Figure 1. STAT Airplane Missions.

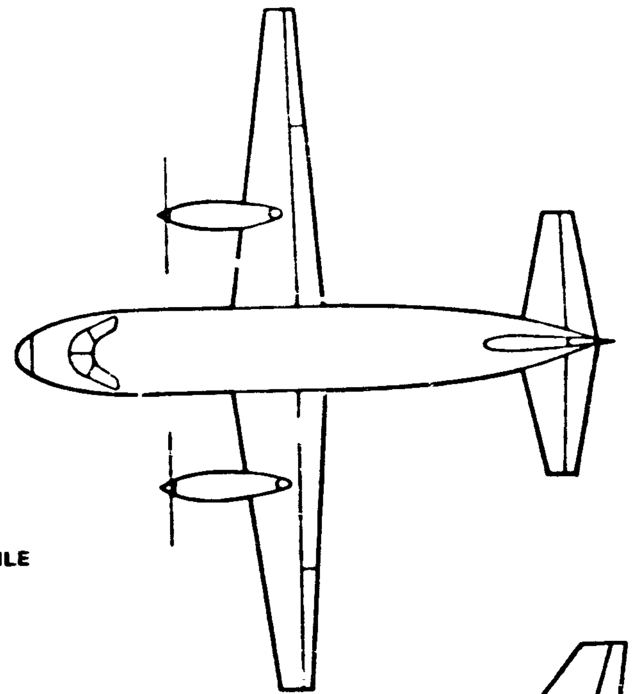
TAKEOFF GROSS WEIGHT = 13,·48 kg (30,308 LB)

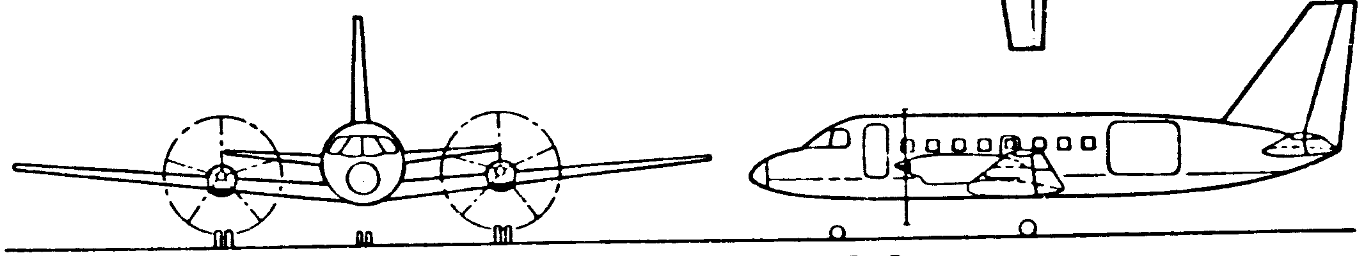
PROPULSION DATA

- SHP (SLS) = 1374 kW (1842 HP)/ENGINE
- STATIC THRUST/WEIGHT RATIO = 0.414
- 5-BLADE PROPELLER
  - MAX TIP SPEED = 220 m/s (722 FT/SEC)
  - ACTIVITY FACTOR/BLADE = 116.0
  - PROP DIAMETER = 3.86 m (12.65 FT)

PERFORMANCE DATA

- CRUISE MACH = 0.45 @ 3650 m (12,000 FT)
- RANGE = 100 NMI
- BLOCK FUEL = 309 kg (681 LB)
- BLOCK TIME = 0.54 HR
- DIRECT OPERATING COST = 8.4065¢/SEAT STATUTE MILE





FUSELAGE:

DIAMETER = 2,74 m (9 FT)
LENGTH = 18.8 m (61.8 FT)

WING:

SPAN = 22.3 m (73.2 FT)
AREA = 46.9 m^2 (505.1 FT^2)
ASPECT RATIO = 10.6

NACELLE:

LENGTH = 4.1 m (13.4 FT)
MEAN DIAMETER = 1.1 m (3.7 FT)
LOCATION = 4.5 m (14.6 FT) FROM AIRCRAFT C/L

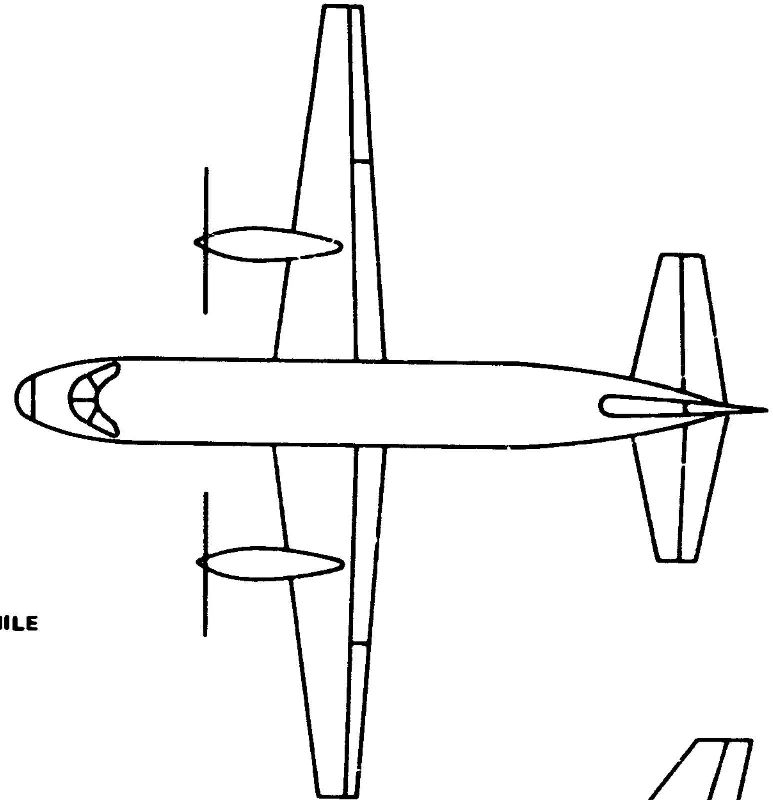
HORIZONTAL TAIL:

SPAN = 8.7 m (28.5 FT)
AREA = 15.2 m^2 (163.0 FT^2)
ASPECT RATIO = 5.0

VERTICAL TAIL:

SPAN = 4.9 m (16.0 FT)
AREA = 14.8 m^2 (159.1 FT^2)
ASPECT RATIO = 1.6

Figure 2. STAT 30-Passenger Airplane.



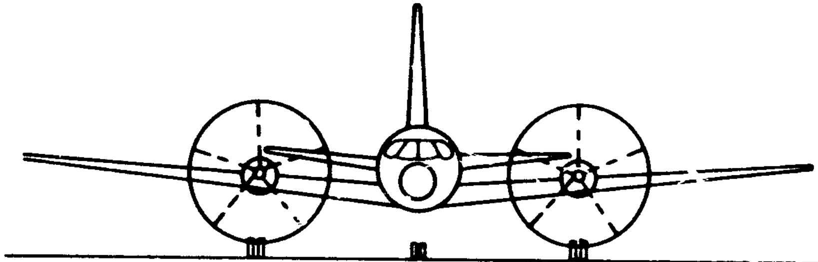
TAKEOFF GROSS WEIGHT = 19,430 kg (42,845 LB)

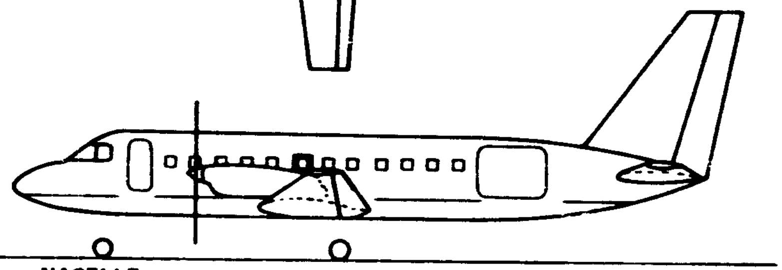
PROPULSION DATA

- SHP (SLS) = 1811 kW (2429 HP)/ENGINE
- STATIC THRUST/WEIGHT RATIO = 0.424
- 5-BLADE PROPELLER
  - MAX TIP SPEED = 215.8 m/s (708 FT/SEC)
  - ACTIVITY FACTOR/BLADE = 115.9
  - PROP DIAMETER = 4.85 m (15.89 FT)

PERFORMANCE DATA

- CRUISE MACH = 0.45 @ 3650 m (12,000 FT)
- RANGE = 100 NMI
- BLOCK FUEL = 411 kg (906 LB)
- BLOCK TIME = 0.45 HR
- DIRECT OPERATING COST = 6.814¢/SEAT STATUTE MILE





FUSELAGE:

DIAMETER = 2.74 m (9 FT)
LENGTH = 22.8 m (74.8 FT)

WING:

SPAN = 26.0 m (85.4 FT)
AREA = 66.3 m^2 (714.1 FT^2)
ASPECT RATIO = 10.2

NACELLE:

LENGTH = 4.6 m (15.2 FT)
MEAN DIAMETER = 1.29 m (4.22 FT)
LOCATION = 5.2 m (17.2 FT) FROM AIRCRAFT C/L

HORIZONTAL TAIL:

SPAN = 11.8 m (38.6 FT)
AREA = 27.7 m^2 (298.1 FT^2)
ASPECT RATIO = 5.0

VERTICAL TAIL:

SPAN = 5.8 m (18.9 FT)
AREA = 14.8 m^2 (159.1 FT^2)
ASPECT RATIO = 1.6

Figure 3. STAT 50-Passenger Airplane.

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TABLE III. GARRETT STAT AIRPLANE SENSITIVITIES.

30-PAX		%Δ DOC - 100NM	%Δ DOC - 600 NM	%Δ Block Fuel - 100 NM	%Δ Block Fuel - 600 NM	%Δ TOGW	%Δ Empty Weight	%Δ Airplane Acquisition Cost
%Δ SFC	+10%	+4.59	+4.31	+11.06	+10.3	+2.0	+1.0	+0.9
	-10%	-4.59	-4.31	-11.06	-10.3	-2.0	-1.0	-0.9
%Δ Engine Weight	+10%	+0.743	+0.598	+1.001	+0.609	+0.979	+1.335	+1.267
	-10%	-0.472	-0.394	-0.413	-0.300	-0.761	-1.109	-1.023
%Δ Engine Acquisition Cost	+10%	+0.283	+0.279	-	-	-	-	+1.261
	-10%	-0.282	-0.279	-	-	-	-	-1.277
%Δ Engine Maintenance %	+10%	+1.110	+1.190	-	-	-	-	-
	-10%	-1.110	-1.190	-	-	-	-	-
50-PAX								
%Δ SFC	+10%	+4.0	+4.34	+10.9	+10.6	+1.88	+0.875	+0.9
	-10%	-4.0	-4.34	-10.9	-10.6	-1.88	-0.875	-0.8
%Δ Engine Weight	+10%	+0.644	+0.518	+0.881	+0.536	+0.897	+1.223	+1.126
	-10%	-0.421	-0.351	-0.314	-0.231	-0.707	-1.030	-0.909
%Δ Engine Acquisition Cost	+10%	+0.236	+0.233	-	-	-	-	+1.311
	-10%	-0.335	-0.331	-	-	-	-	-1.311
%Δ Engine Maintenance %	+10%	+0.819	+1.076	-	-	-	-	-
	-10%	-0.819	-1.076	-	-	-	-	-

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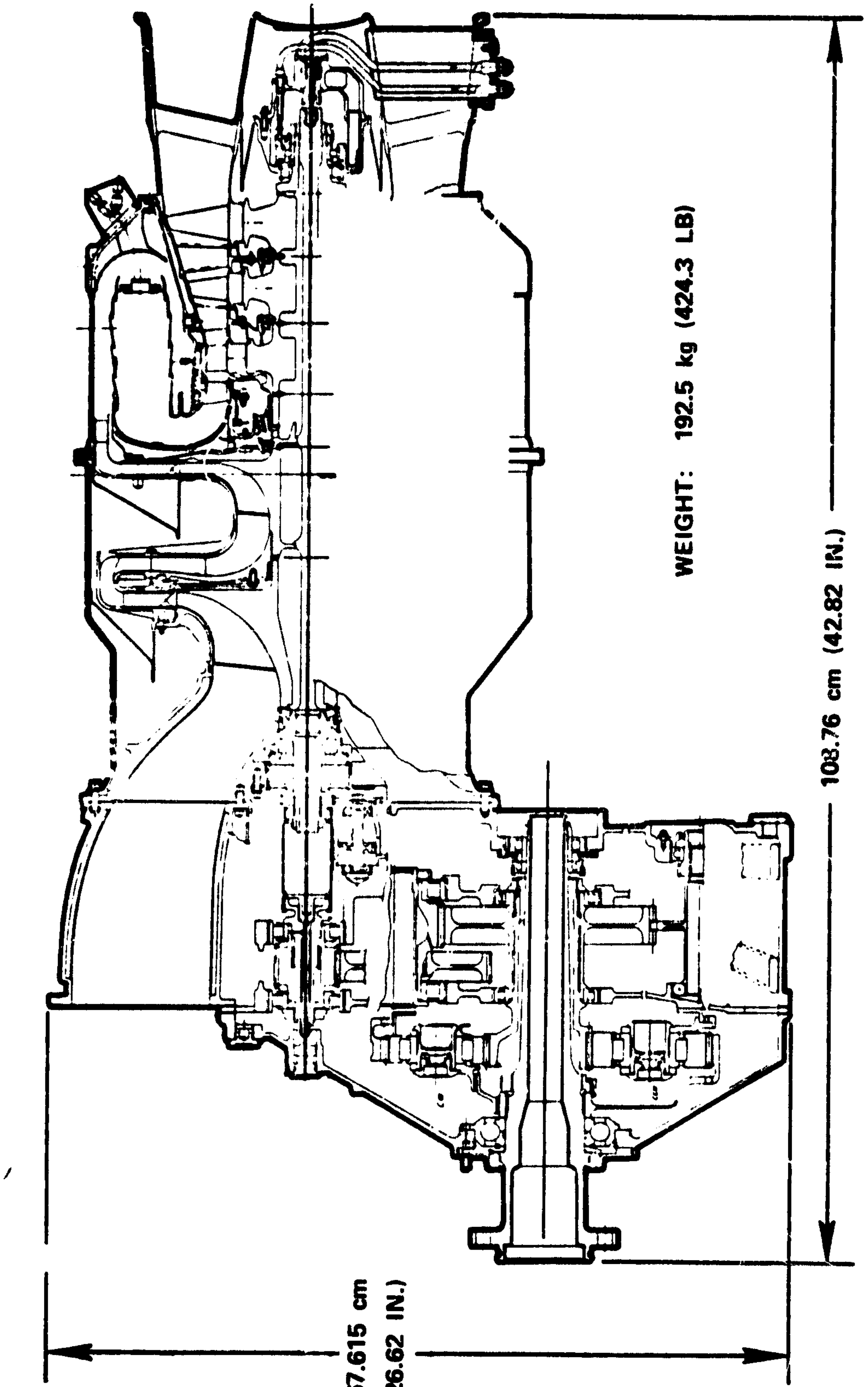

Figure 4. 1980-Technology Engine.

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TABLE IV. STAT 1980 BASELINE ENGINE PERFORMANCE (UNINSTALLED).

			Takeoff Power Sea-Level Static [288K (59°F)]		Cruise at 3650 m (12,000 Ft) for ISA, M=0.452	
Parameter	SI UNIT	Customary Unit	SI Value	Cust. Value	SI Value	Cust. Value
Shaft Horsepower	kw	hp	745.7	1000	521.2	699
BSFC	kg/kW·hr)	(lbm/hr)/hp	0.3394	0.558	0.3358	0.552
Net Jet Thrust	N	lbf	502.6	113	-4.4	-1
P/P (Compressor)	--	--	10.6	10.6	9.8	9.8

performance characteristics plus weight and cost values were furnished to NASA-Ames for incorporation in their airplane synthesis program. The results of that incorporation are reflected in the airplane design data given in Appendix I. The characteristics of the Model TPE331-11 were scaled to the horsepower requirements of the 30- and 50-passenger airplanes.

The 1985 derivative engine shown in Figure 5 was based on a Garrett Model TPE331 growth study engine. The basic performance characteristics of the 1985 derivative engine are given in Table V. These performance characteristics were also scaled to the horsepower requirements of the 30- and 50-passenger airplanes.

Selection of the 1990 STAT technology engine was based on the results of a cycle parametric study, as well as considerations of weight, cost, fabrication complexity, and maintainability. For this engine, only concentric-shaft, front-drive, free-turbine engines were considered. Single-shaft and low-spool drive (compressor booster stages on the output shaft) were considered later in the program and are subsequently discussed. All engines in the initial parametric study consisted of two-stage, centrifugal-compressor configurations driven by single- or two-stage, cooled, axial, gas-generator turbines, a reverse-flow annnular combustor, and a two-stage, uncooled low-pressure (LP) or power turbine. All engines had a constant inlet corrected flow rate of 4.54 kg/s (10 lbm/sec) at the cruise design condition. The parameters that were varied were overall compression ratio (OCR) and turbine rotor inlet temperature (TRIT). The schedule of efficiency versus pressure ratio used for the two-stage centrifugal compressor and the correlation of turbine efficiency with turbine work used for the gas generator and power turbines are shown in Figure 6.

The results of the cycle parametric study are shown in Figure 7 in terms of thrust (propeller and jet) specific fuel consumption (TSFC) and specific thrust, which is defined as the ratio of propeller plus jet thrust to the inlet airflow rate. A propeller having an efficiency of 0.842 was assumed. As indicated by the dashed line in Figure 7, a fuel consumption penalty is incurred if a single-stage, high-pressure (HP) turbine is used. The flight condition at 5181.6 meters (17,000 feet) was chosen as a compromise between the 100-nmi and 600-nmi cruise altitudes of 3658 and 6096 meters, respectively.

The cycle parametric showed that a TRIT of 2350°F was very near optimum for all pressure ratios investigated and pressure ratios between 16 and 20:1 yielded a TSFC variation of less than 1 percent. Weight, cost and complexity generally benefit from selection of a lower pressure ratio for small differences in SFC. Accordingly, an OCR of 16:1 and a TRIT of 2350°F were selected for the 1990 engine cycle.

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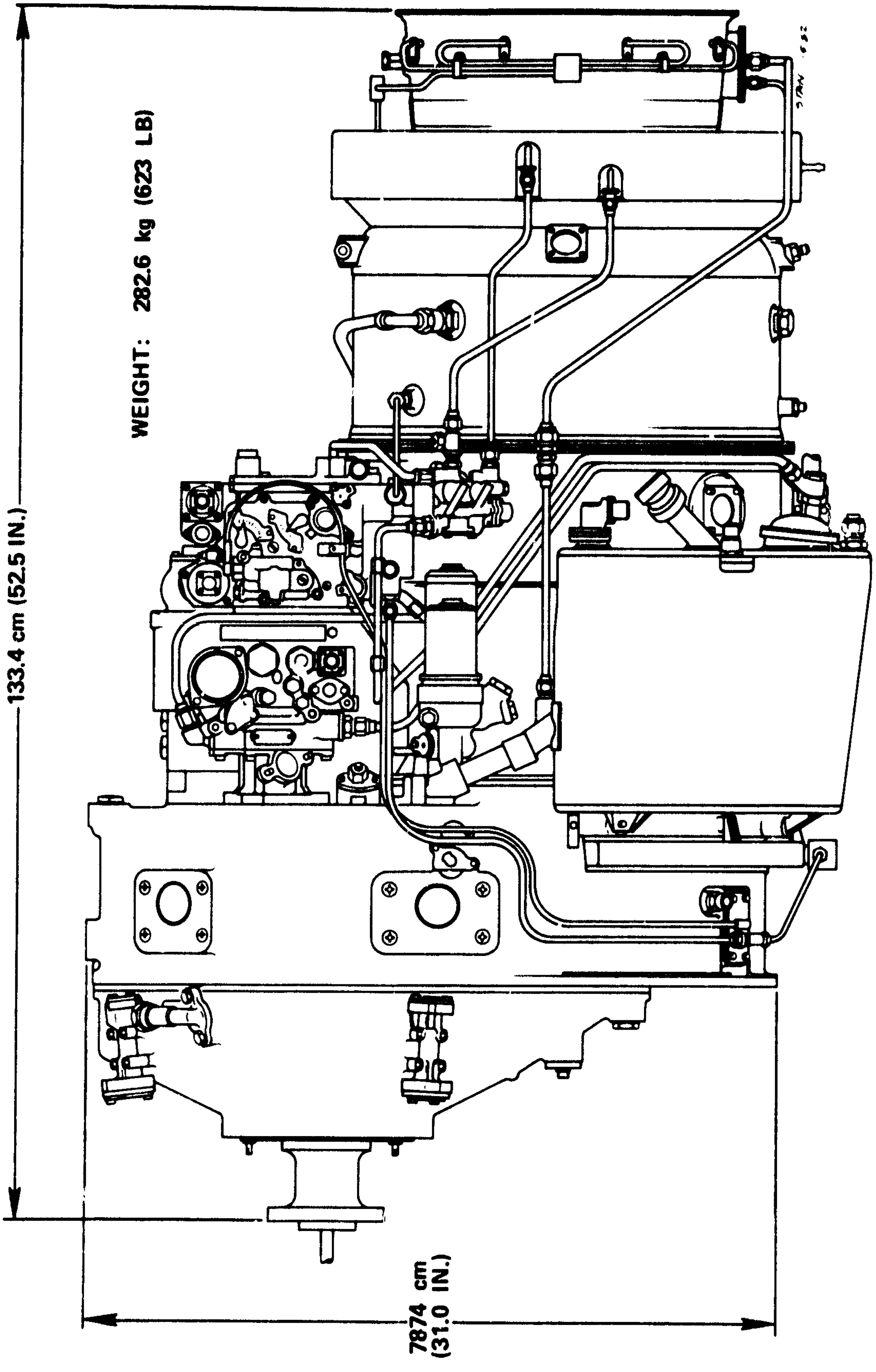

Figure 5. 1985 Derivative Engine.

TABLE V. STAT 1985 DERIVATIVE ENGINE PERFORMANCE (UNINSTALLED)

			Takeoff Power Sea-Level Static [288 K (59°F)]		Cruise at 3650 m (12,000 Ft) for ISA, M=0.45	
Parameter	SI UNIT	Customary Unit	SI Value	Cust. Value	SI Value	Cust. Value
Shaft Horsepower SHP	kW	hp	1226.7	1645	850.8	1141
BSFC	kg/kW·hr	(lbm/hr)/hp	0.3054	0.502	0.2975	0.489
Net Jet Thrust	N	lbf	742.9	167	-8.9	-2
P/P (Compressor)	--	--	10.8	10.8	10.0	10.0

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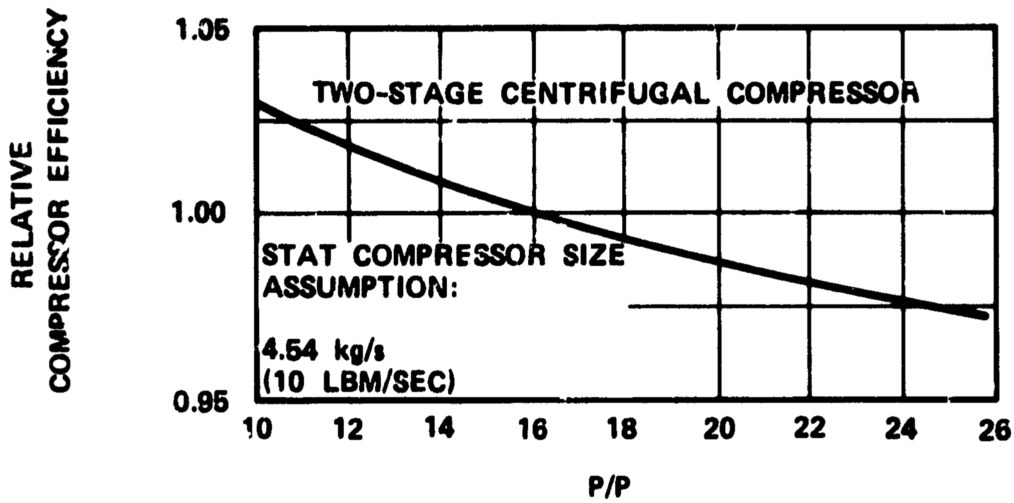

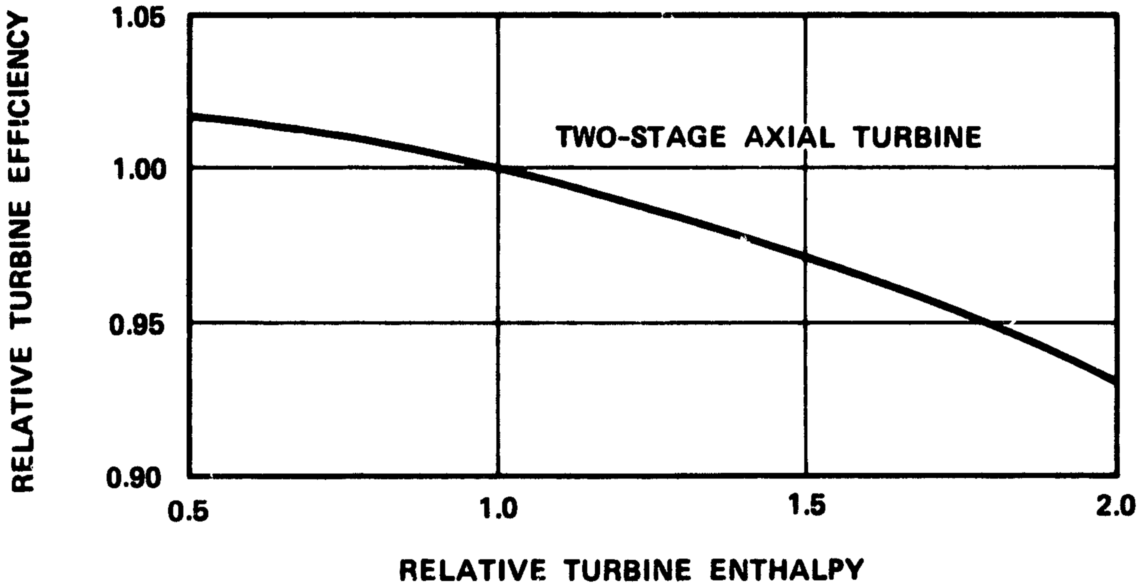

Figure 6. Relative Component Efficiency Schedules.

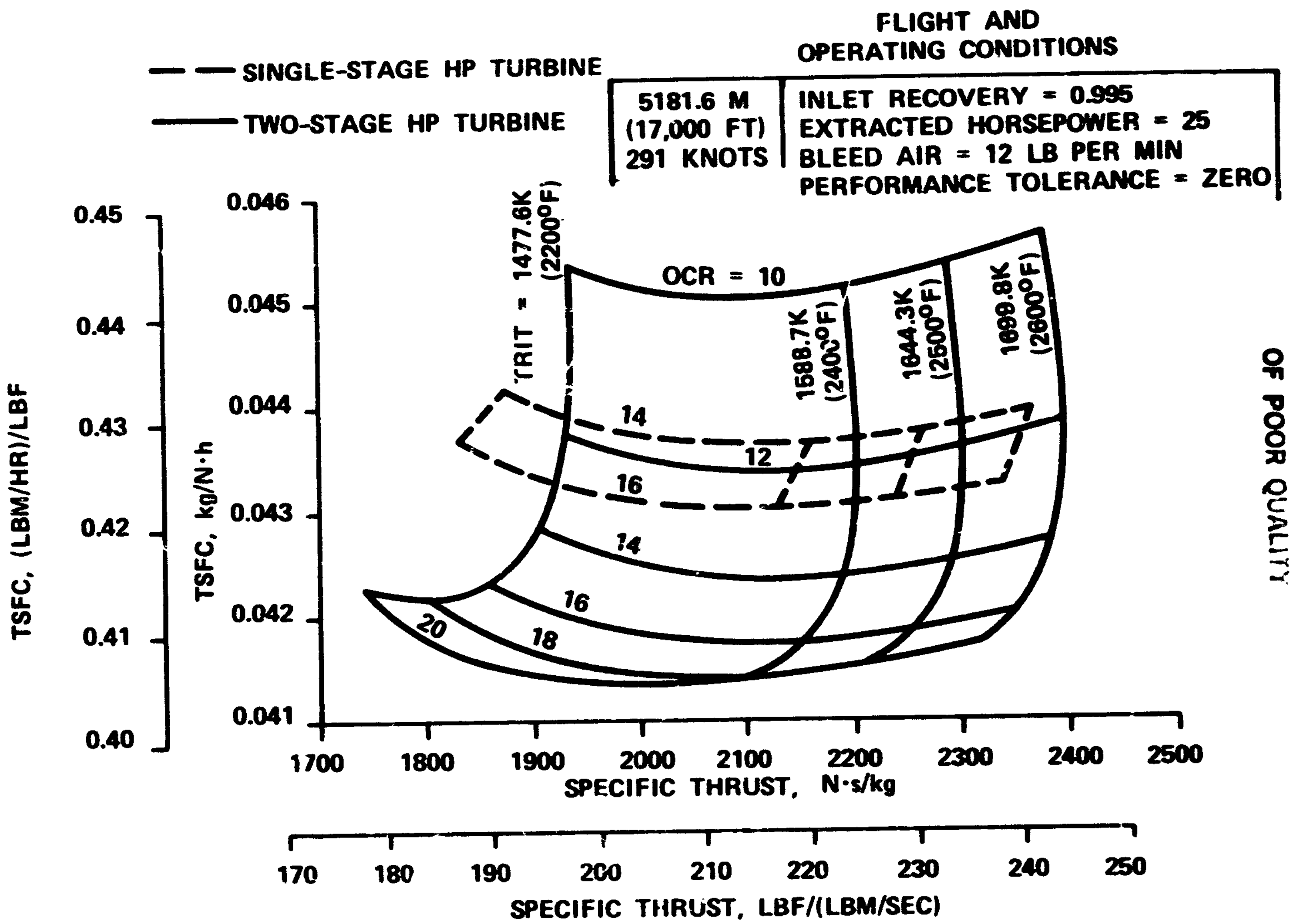

Figure 7. STAT Cycle Parametric Study Results.

The parametric analysis led to the definition of the 1990 STAT engine shown in Figure 8. The basic performance and cycle characteristics for this engine based on off-design analysis are given in Table VI for the sea-level takeoff and 3650-m (12,000-ft) cruise conditions.

The performance, weight, cost, and maintenance characteristics of these three engines were used with the airplane sensitivities to compute the operating costs and benefits for the STAT baseline airplane. Computation of the costs was based on equations furnished by NASA-Ames (as shown in Table VII) with two exceptions. The engine maintenance costs were based on Garrett experience with TPE331 engines in commuter service and on predictions for derivative and advanced engines. The differences in maintenance costs, shown in Table VI between the 1980 baseline, the 1985 derivative, and the 1990 STAT engines are due principally to the design philosophy used for the engines. The 1980 baseline engine was originally designed for business or executive aircraft, with the majority of maintenance activities to be performed at factory-authorized repair and overhaul centers. The 1985 and 1990 engines were designed principally for commuter transport applications, with the majority of maintenance activities, including major module replacement, to be performed by the operators. As discussed subsequently, modular construction is the principal contributor to the reduction in maintenance cost. The maintenance burden was computed only for the airframe, since the engine maintenance burden was included with the engine maintenance cost. Propeller maintenance costs were initially based on the on-condition maintenance cost factors furnished by Hamilton Standard in propeller data packages prepared under Contract NAS3-22039. These factors were subsequently revised, as discussed later, however the overall effect on DOC was not more than 0.1 percent. The original and revised factors are given in Table VIII. It should be noted that all costs referred to throughout this document are in constant 1980 dollars.

The benefits of the 1985 derivative engine and the 1990 baseline engine relative to the 1980 baseline engine for the 30- and 50-passenger airplanes are shown in Tables IX and X, in terms of the 100-nmi DOC. These benefits were estimated using the sensitivities developed for the baseline aircraft and do not include any benefits due to advanced airframe technology. The 50-passenger design used a scaled version as the 30-passenger engine, and no adjustments were made for Reynolds number, clearance or other size related effects. The 1985 derivative engine resulted in a decrease in DOC of more than 10 percent for both the 30- and 50-passenger designs; and the 1990 technology engine resulted in a DOC reduction of more than 15 percent relative to the 1980 technology configuration. Over half of the 15-percent improvement in DOC due to the 1990 technology engine was the result of improvements in BSFC. The improvement in BSFC is attributable primarily

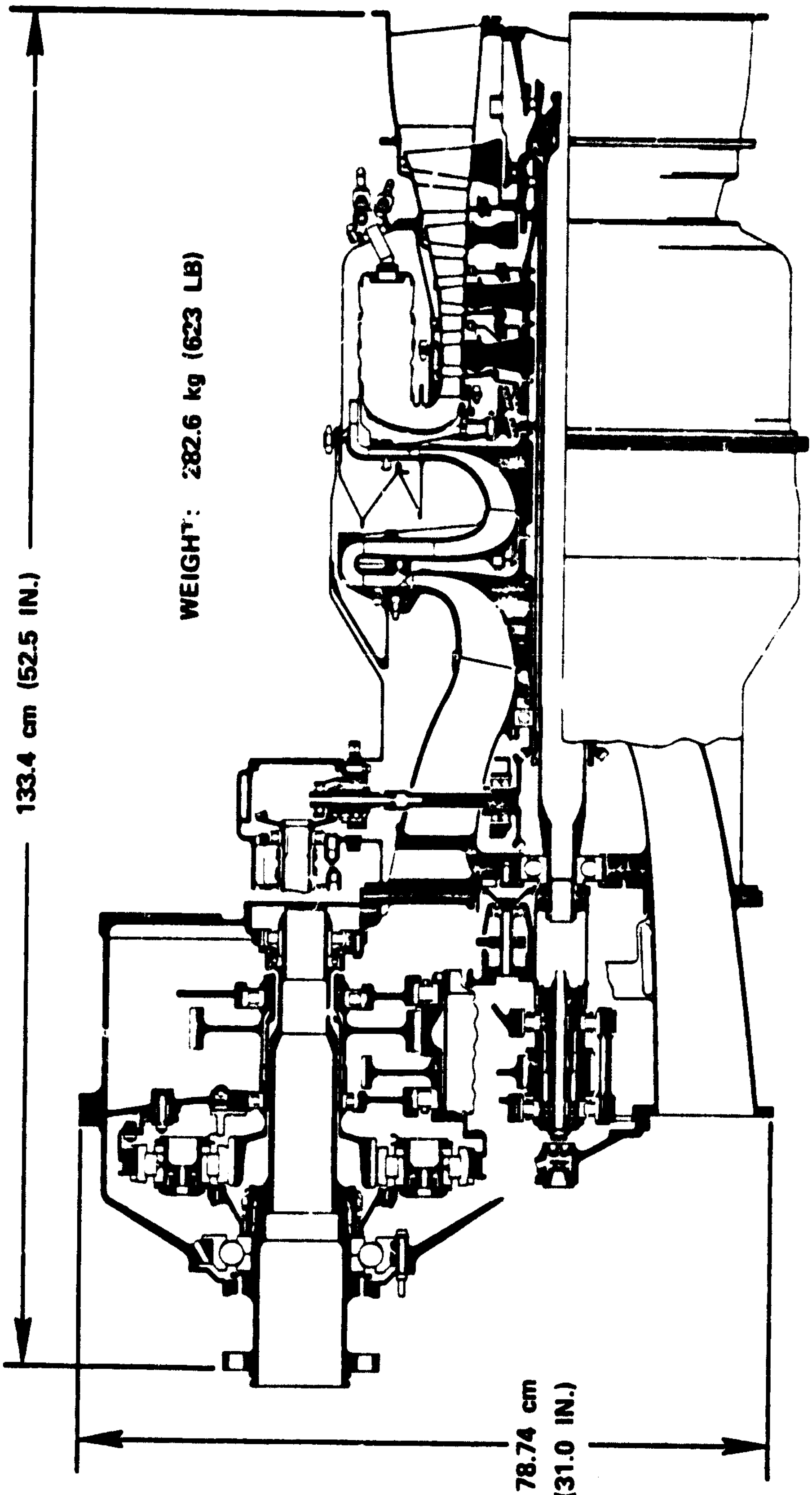

Figure 8. 1990 Stat Engine.

TABLE VI. 1990 STAT ENGINE PERFORMANCE (UNINSTALLED).

			Takeoff Power Sea-Level Static [288 K (59°F)]		Cruise at 3650 m (12,000 Ft) for ISA, M=0.452	
Parameter	SI UNIT	Customary Unit	SI Value	Cust Value	SI Value	Cust Value
Shaft Horsepower	kW	hp	1343.8	1802	1190.9	1597
BSFC	kg/kW·hr	(lbm/hr)/hp	0.2689	0.442	0.2391	0.393
Net Jet Thrust	N	lbf	453.7	102	-57.8	-13
P/P (Compressor)	--	--	14	14	15.2	15.2
Turbine Inlet Temperature	K	°F	1644.3	2500	1560.9	2350

TABLE VII. STAT OPERATING COST EQUATIONS (FURNISHED BY NASA-AMES).

Utilization = 2800 hr/yr　　Insurance rate = 1.5 percent

Depreciation period = 12 yr　　Residual value = 15 percent

DIRECT OPERATING COSTS (DOC), $/BLOCK-HOUR

(1) Flight crew = 2.5 x (max passenger seats)

(2) Fuel, oil, taxes = 0.156 x ($/gal x (block fuel/block time))

(3) Insurance = (5.357×10^{-6}) x (total aircraft cost)

(4) Airframe maintenance:

Labor = $0.0115 \times (W_{AF}{}^{0.575}) \times$ (rate/hr)

Material = $0.115 \times (W_{AF}{}^{0.575})$

= $0.0115\ (W_{AF}{}^{0.575}) \times$ (rate/hr + 10)

(5) Engine maintenance:*

Mission Length, nmi	600		100	
Airplane Pax	30	50	30	50
1980 Baseline	47.64	56.70	63.64	75.58
1985 Derivative	20.80	25.61	27.78	34.14
1990 STAT Technology	20.05	24.61	26.79	32.79

(6) Propeller maintenance = F(blades, TBO) per Hamilton Standard

(7) Maintenance burden = 0.80 x (airframe labor + engine labor)

(8) Depreciation = (2.53×10^{-5}) x (total aircraft cost) x (spare factor)

where: spare factor = 1.08 for 30 seats

spare factor = 1.12 for 50 seats

INDIRECT OPERATING COSTS (IOC), $/BLOCK-HOUR

IOC = 0.80 x DOC + flight attendant cost

where: flight-attendant cost = $14.40/block-hour for 20 < seats < 50

*Garrett model based on TPE331

TABLE VIII. PROPELLER ON-CONDITION MAINTENANCE COST IN $/FLT-HR PER $1000 ACQUISITION COST.

	ORIGINAL*	REVISED
Current Technology	0.0153	0.036
Advanced Technology, 3-way	0.0320	0.027
Advanced Technology, 4-way	0.0360	0.030
Advanced Technology, 6-way	0.0460	0.036

*Hamilton-Standard propeller data packages prepared under Contract NAS3-22039

TABLE IX. STAT ENGINE BENEFITS BASED ON 100-nmi MISSION FOR 30-PASSENGER AIRPLANE.

Parameters*	SI Units	Customary Units	1980 Baseline Engine		1985 Derivative Engine		1990 STAT Engine	
			SI Values	Cust. Values	SI Values	Cust. Values	SI Values	Cust. Values
Shaft Horsepower	kW	hp	1373.6	1842	1359.4	1823	1343.8	1802
BSFC	kg/kW·hr	(lbm/hr)/hp	0.3394	0.558	0.3054	0.502	0.2689	0.442
Weight	kg	lbm	345.2	761	295.7	652	282.6	623
Cost	k$	k$	230.0	230.0	242.5	242.5	241.0	241.0
Δ% DOC from Baseline								
SFC	%	%	--	--	-4.5	-4.5	-9.3	-9.3
Weight	%	%	--	--	-0.6	-0.6	-0.7	-0.7
Cost	%	%	--	--	+0.2	+0.2	+0.2	+0.2
Maintenance Cost	%	%	--	--	-6.7	-6.7	-6.9	-6.9
ΣΔ% DOC	%	%	--	--	-11.6	-11.6	-16.7	-16.7
Relative DOC	--	--	--	--	0.844	0.844	0.833	0.833

*Engine characteristics at sea level static, standard day, takeoff power uninstalled as sized for the aircraft.

TABLE X. STAT ENGINE BENEFITS BASED ON 100-nmi MISSION FOR 50-PASSENGER AIRPLANE.

Parameters*	SI Units	Customary Units	1980 Baseline Engine		1985 Derivative Engine		1990 STAT Engine	
			SI Values	Cust. Values	SI Values	Cust. Values	SI Values	Cust. Values
Shaft Horsepower	kW	hp	1811.3	2429	1795.6	2408	1777.7	2384
BSFC	kg/kW·hr	(lbm/hr)/hp	0.3394	0.558	0.3054	0.502	0.2689	0.442
Weight	kg	lbm	520.27	1027	390.54	861	373.76	824
Cost	k$	k$	304.0	304.0	320.3	220.3	318.1	318.1
Δ% DOC from Baseline								
SFC	%	%	--	--	-4.2	-4.2	-8.5	-8.5
Weight	%	%	--	--	-0.8	-0.8	-0.9	-0.9
Cost	%	%	--	--	+0.1	+0.1	+0.1	+0.1
Maintenance Cost	%	%	--	--	-5.7	-5.7	-5.9	-5.9
ΣΔ% DOC	%	%	-	-	-10.6	-10.6	-15.2	-15.2
Relative DOC	--	--	--	--	0.894	0.894	0.848	0.848

*Engine characteristics at sea level static standard day, takeoff power, uninstalled as sized for the aircraft.

to higher pressure ratio and improved component efficiencies. A detailed breakdown of the contribution of cycle, component efficiency and other improvements was not made due to the significant configuration change between the 1990 technology engine (free turbine, concentric shaft) and the 1980 baseline engine, (single shaft).

Almost 40 percent of the DOC improvement is due to the lower maintenance cost of the 1990 technology engine. As stated previously, the majority of the maintenance cost savings is due to the modular engine design.

The DOC improvements resulting from the 1985 derivative engine were split between the improvement in BSFC and maintenance cost. The BSFC improvements were due solely to improvements in component efficiencies as the cycles of the 1980 and 1985 engines are almost identical.

An additional investigation was made to determine what further benefits might result from considering scale effects between the 30- and 50-passenger engines. A scale factor of 1.323 was used, based on the ratio of engine rated horsepowers. It was found that because the actual geometric change was relatively small (approximately 15-percent diameter increase), the combination of clearance and Reynolds-number effects resulted only in a 0.4-point increase in turbine efficiency. Although small, this improvement results in a shaft-horsepower increase of slightly over 0.5 percent at the cruise conditions, and an SFC reduction of slightly over 0.3 percent, which translates to a DOC reduction of about 0.2 percent.

The results of the DOC computations are presented as pie charts in Figures 9 through 14. These charts show the total DOC (in constant 1980 dollars) for the 30- and 50-passenger airplanes, broken down by the elements given in the equations of Table VII for the following conditions:

Fuel Cost, \$/l	Fuel Cost, \$/gal	Engine Technology Base	Mission Length, nmi
0.264	1.00	1980	600 & 100
0.264	1.00	1985	600 & 100
0.264	1.00	1990	600 & 100
0.396	1.50	1980	600 & 100
0.396	1.50	1985	600 & 100
0.396	1.50	1990	600 & 100

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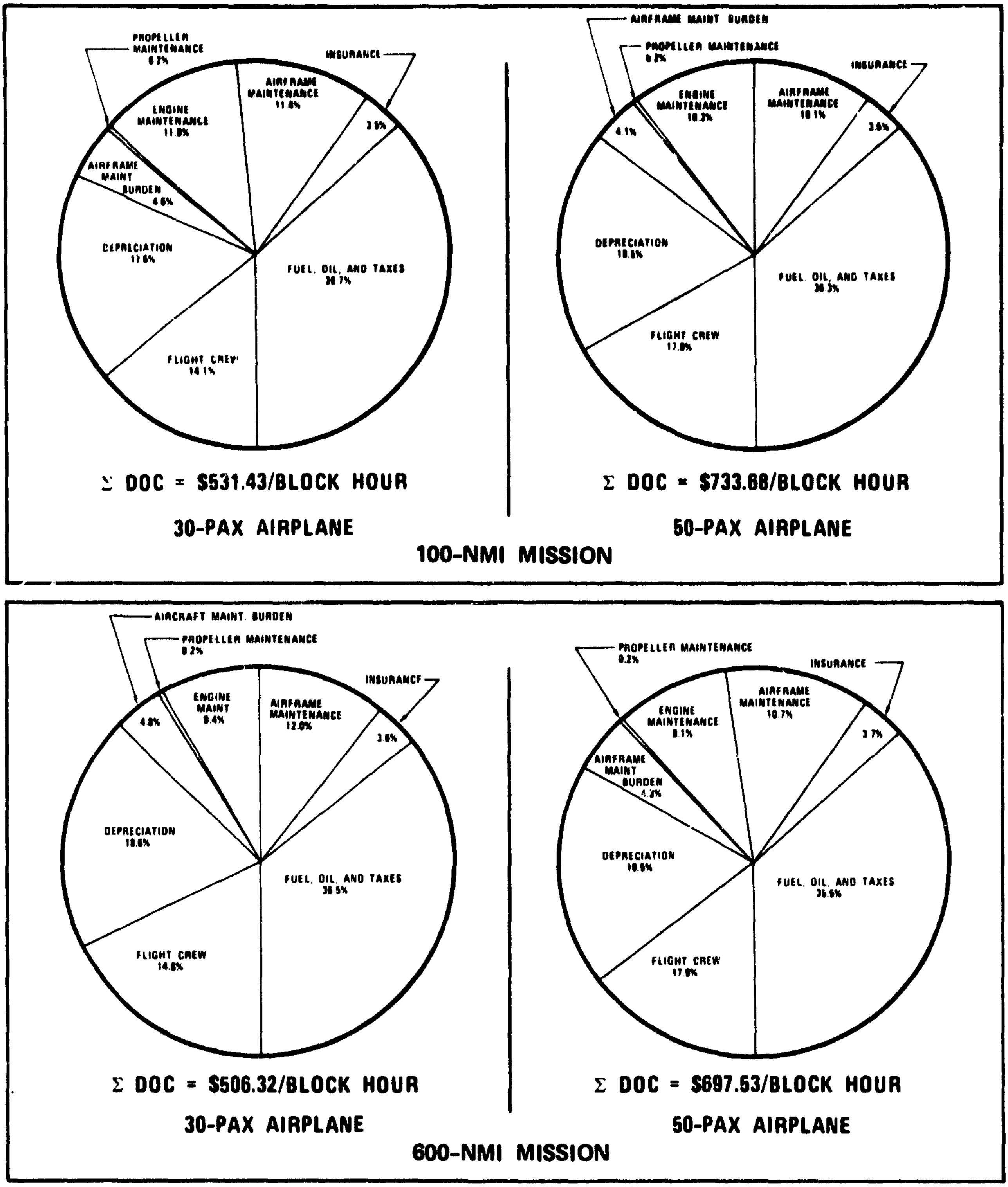


Figure 9. STAT DOC for 1980 Baseline Engine Based on $0.264/Liter ($1.00/Gallon) Fuel Cost.

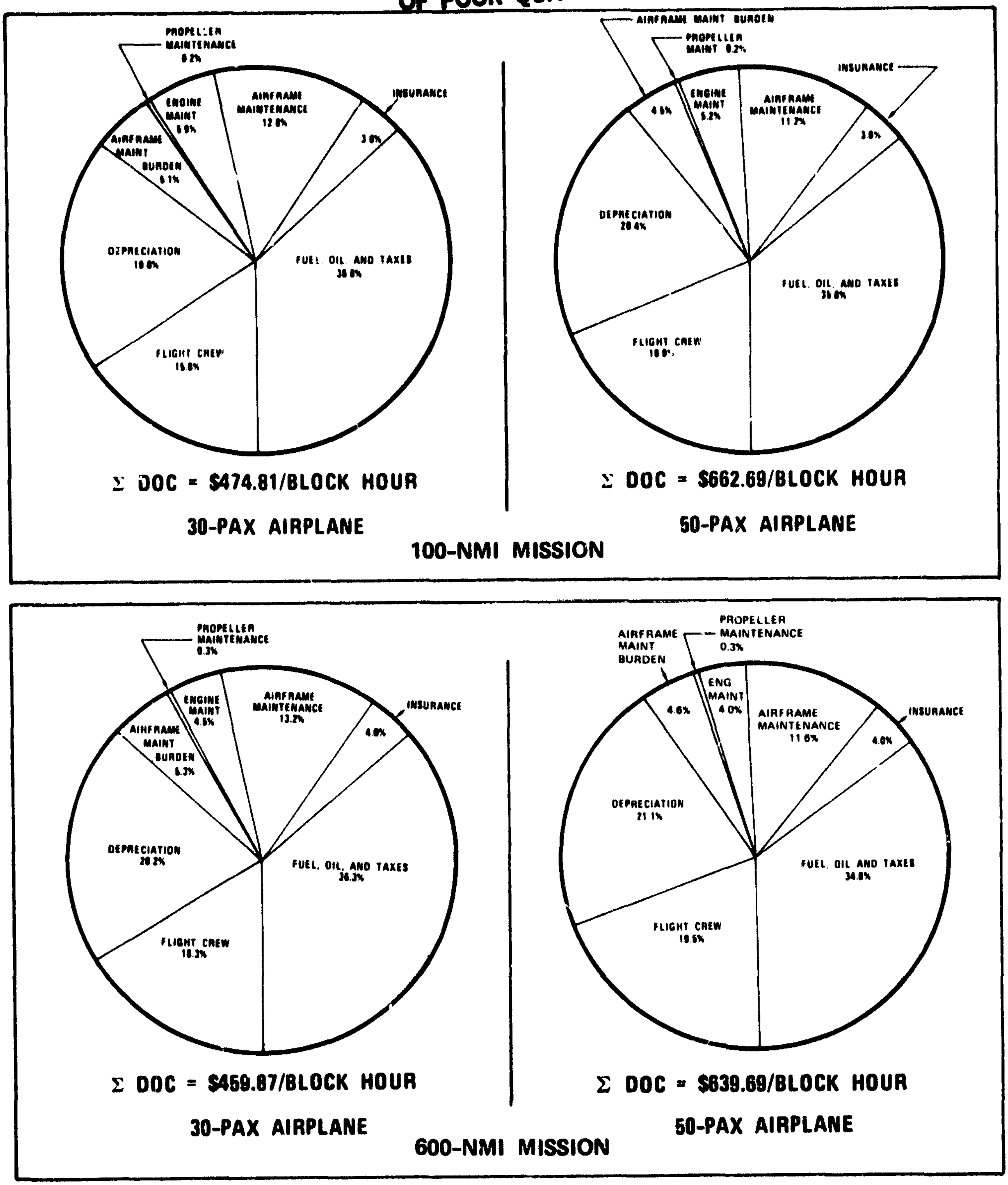


Figure 10. STAT DOC for 1985 Derivative Engine Based on $0.264/Liter ($1.00/Gallon) Fuel Cost.

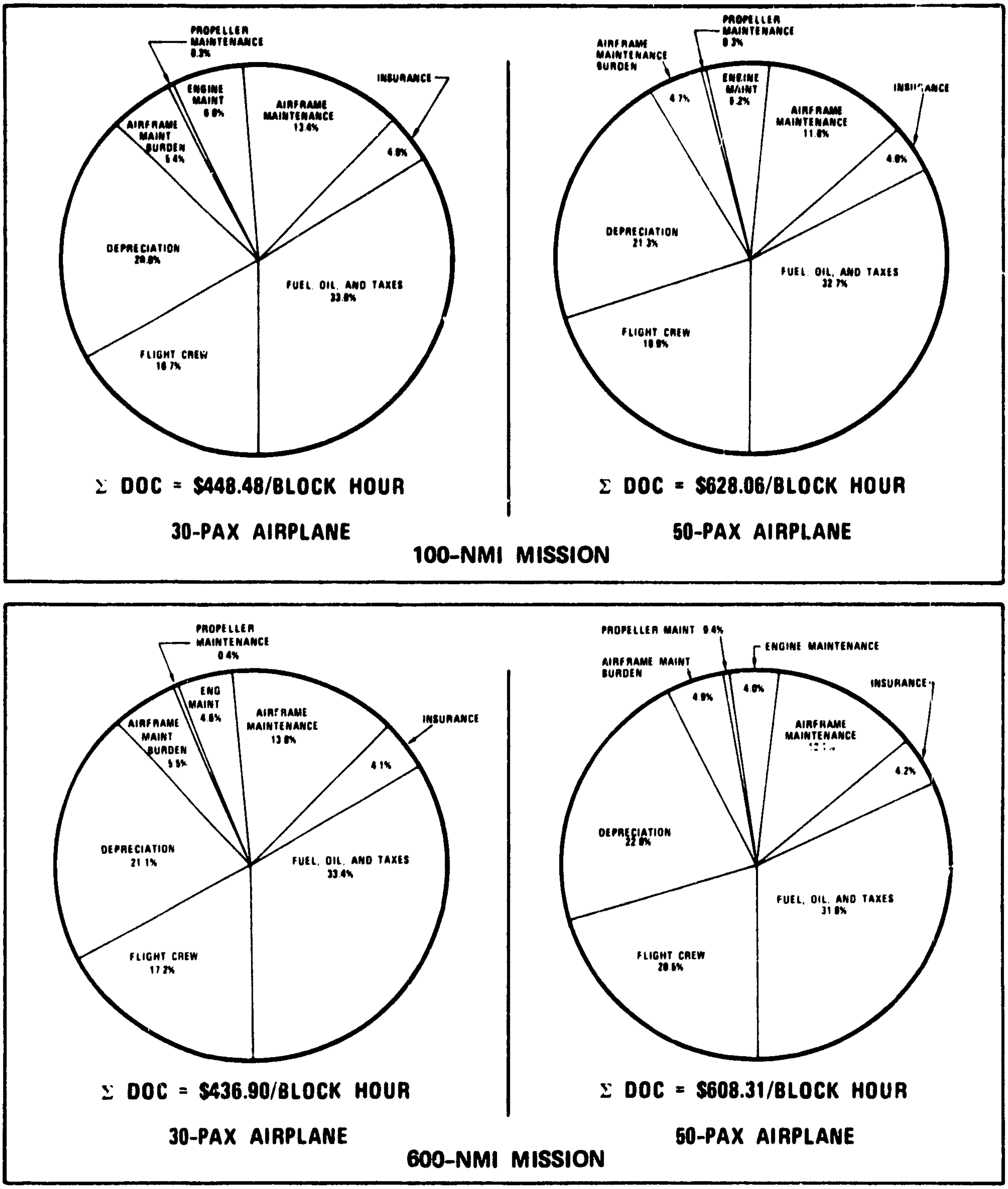


Figure 11. STAT DOC for 1990 STAT Engine Based on $0.264/Liter ($1.00/Gallon) Fuel Cost.

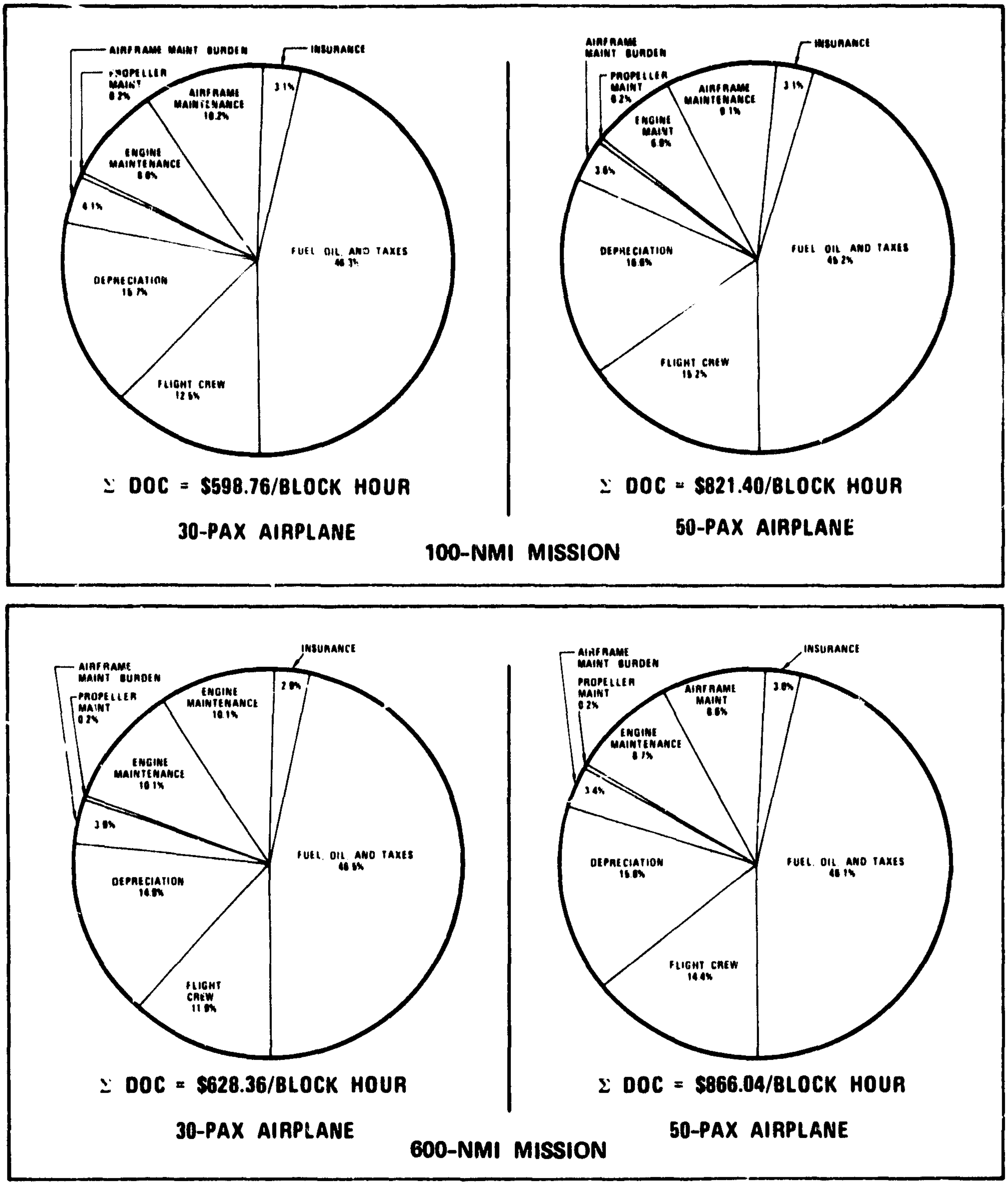


Figure 12. STAT DOC for 1980 Baseline Engine Ease on $0.396/Liter ($1.50/Gallon) Fuel Cost.

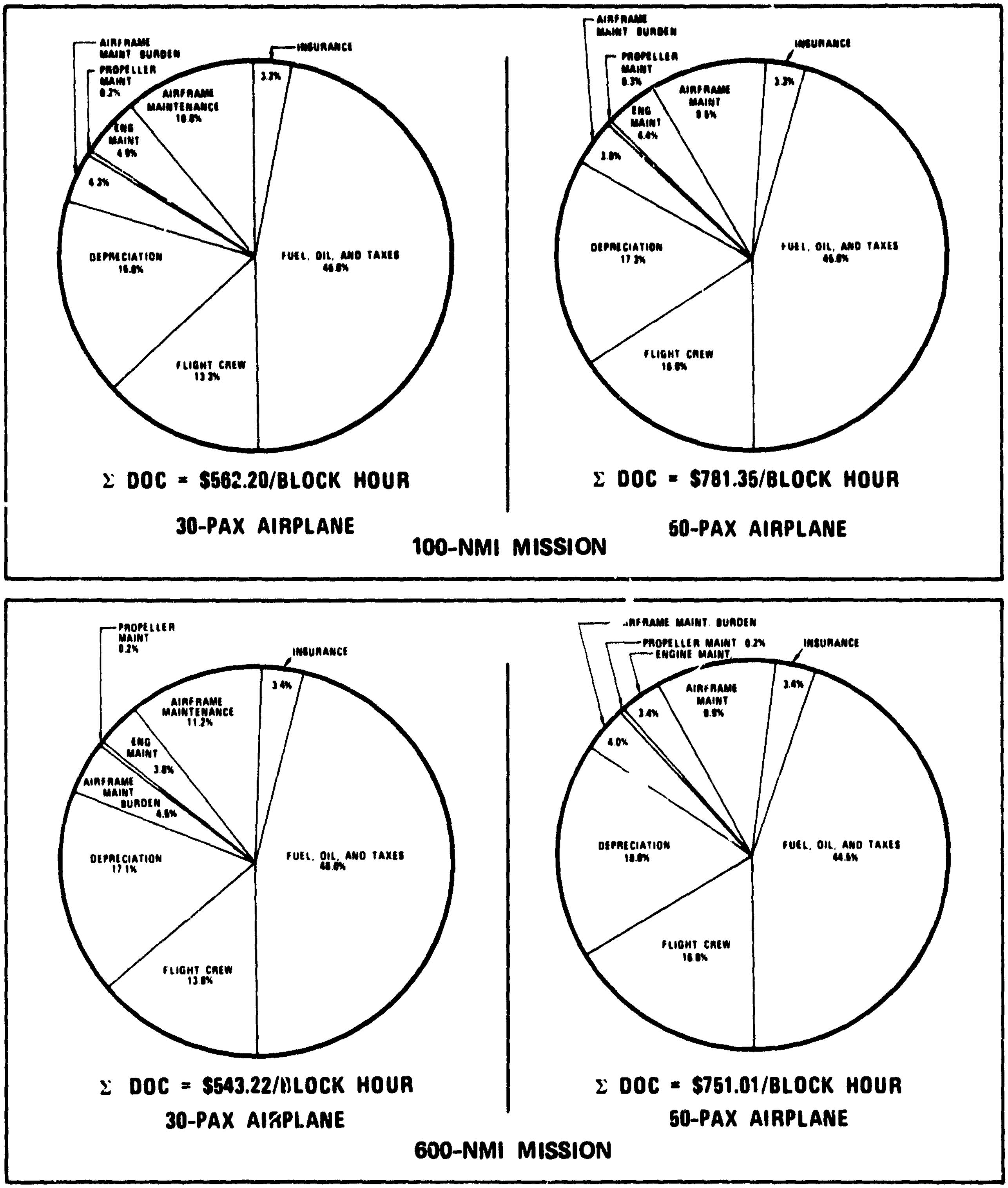


Figure 13. STAT DOC for 1985 Derivative Engine Base on $0.396/Liter ($1.50/Gallon) Fuel Cost.

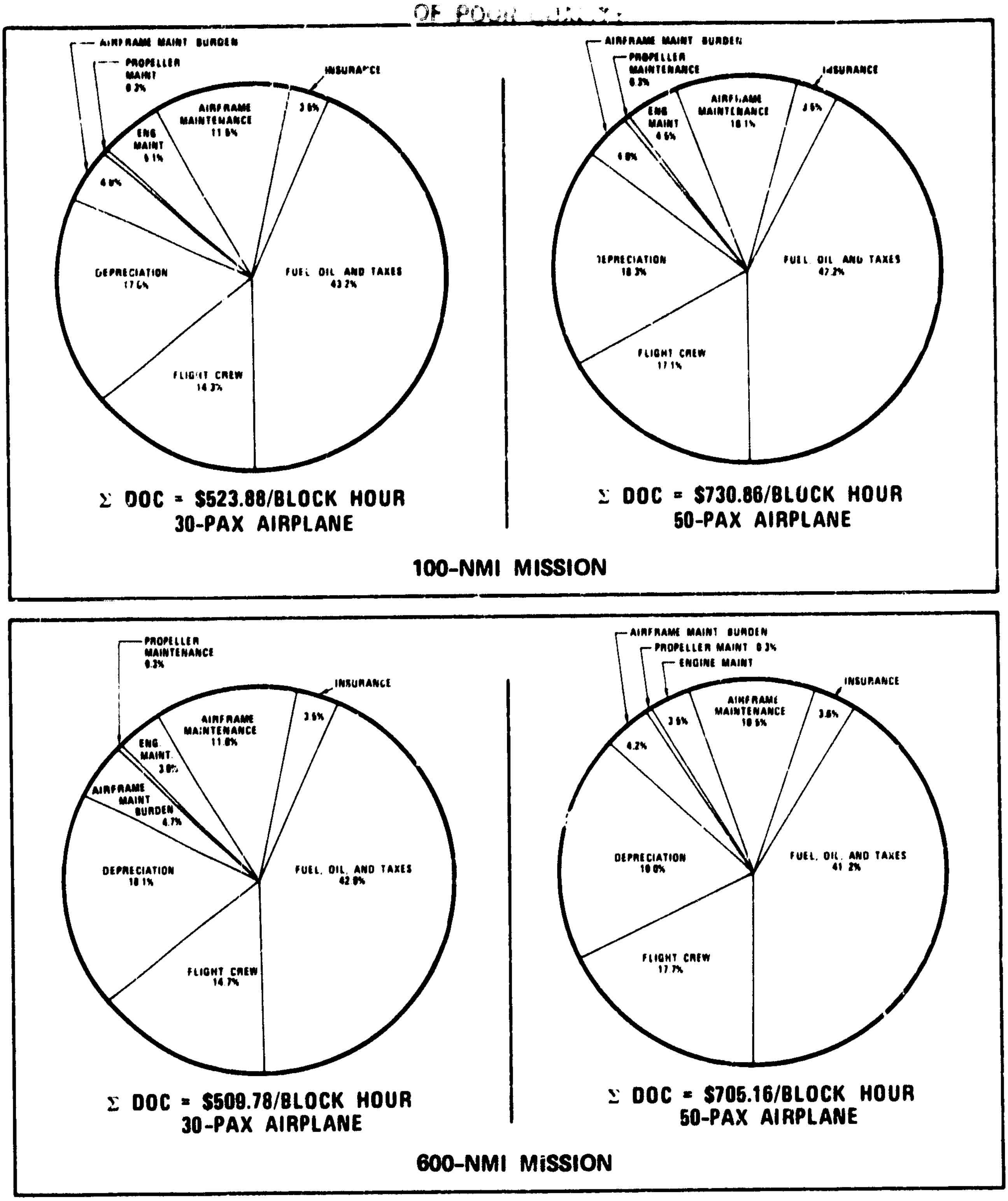


Figure 14. STAT DOC for 1990 STAT Engine Based on $0.396/Liter ($1.50/Gallon) Fuel Cost.

A comparison of these figures reveals several items that warrant comment.

The heavy influence of SFC (block fuel) on DOC is apparent. With the fuel price at $0.264/1 ($1.00/gal), fuel cost represents more than one-third of the total DOC. The strong influence of fuel cost is also apparent. With the fuel price at $0.396/1 ($1.50/gal), fuel cost represents approximately 45 percent of the total DOC, and at $0.528/1 ($2.00/gal), although not shown herein, the fuel cost exceeds 50 percent. The influence of engine technology is presented in Figure 15, which shows that fuel costs decline approximately 22 percent as the engine component efficiencies and cycle quality improves.

It is notable that engine maintenance costs are markedly reduced from the 1980 baseline engine to the 1985 derivative and the 1990 advanced engines, as indicated on Figure 16. This reduction is due to fully modular construction, provisions for on-the-wing maintenance, repairable components, and component design criteria which is consistent with the severe engine duty cycle imposed by commuter operations. The TPE331-11 engine was designed for the business/executive market where small fleet sizes and maintenance organization necessitated return of the complete engine to the factory or overhaul shops for overhaul at time intervals set by the lower durability components. Where larger fleets are maintained, more extensive maintenance organization can be justified and direct maintenance costs can be reduced through modular maintenance where only components requiring maintenance are involved.

Repairability is a significant factor. For example, the 1985 and 1990 engines utilize inserted blades in all turbine stages which allow only damaged blades to be replaced.

The other major factor in reducing the maintenance cost of the 1985 and 1990 engines was improved durability. The 1985 and 1990 engines are designed both for the stress rupture and low-cycle-fatigue requirements of commuter operations.

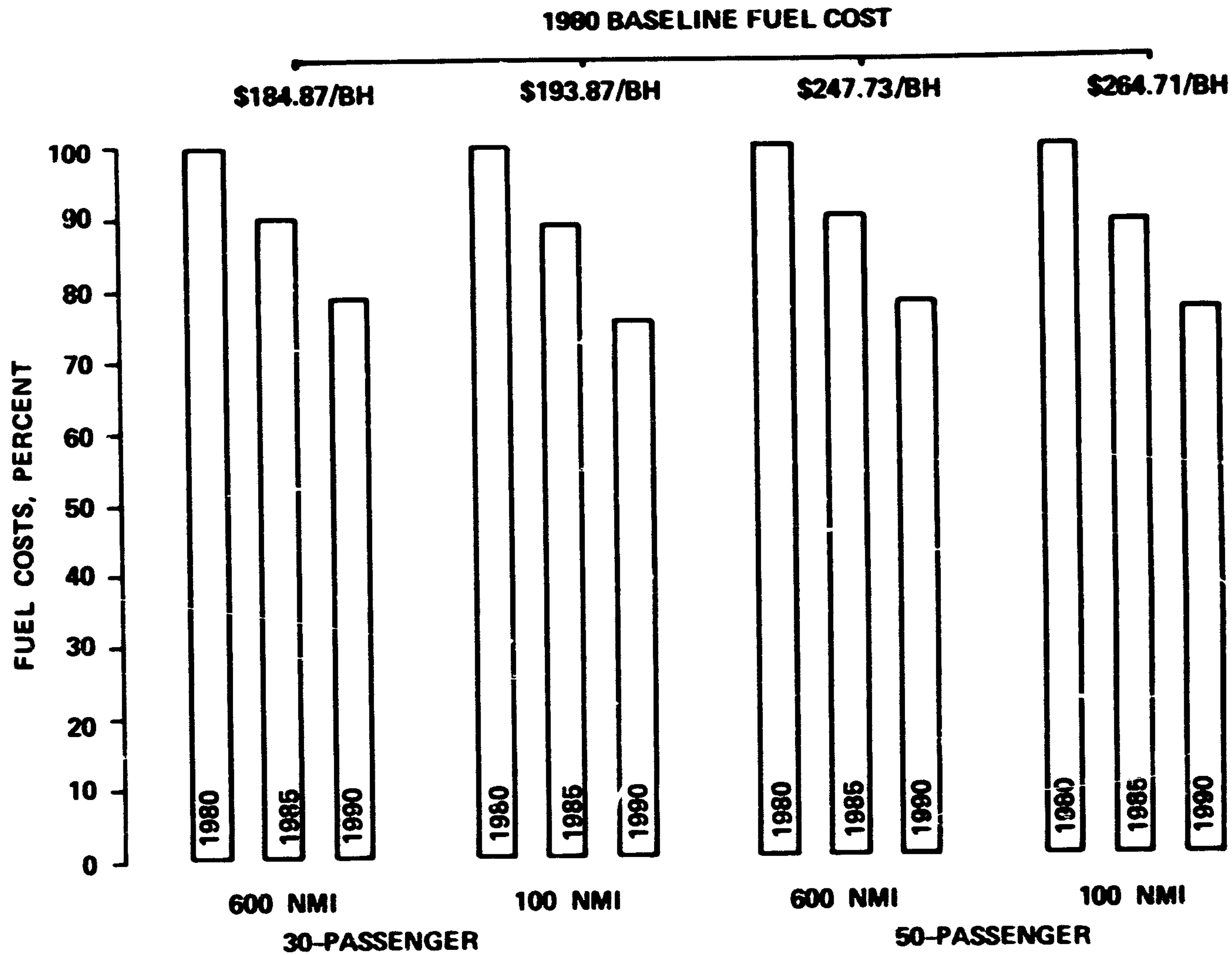


Figure 15. STAT Fuel Costs Relative to 1980 Baseline.

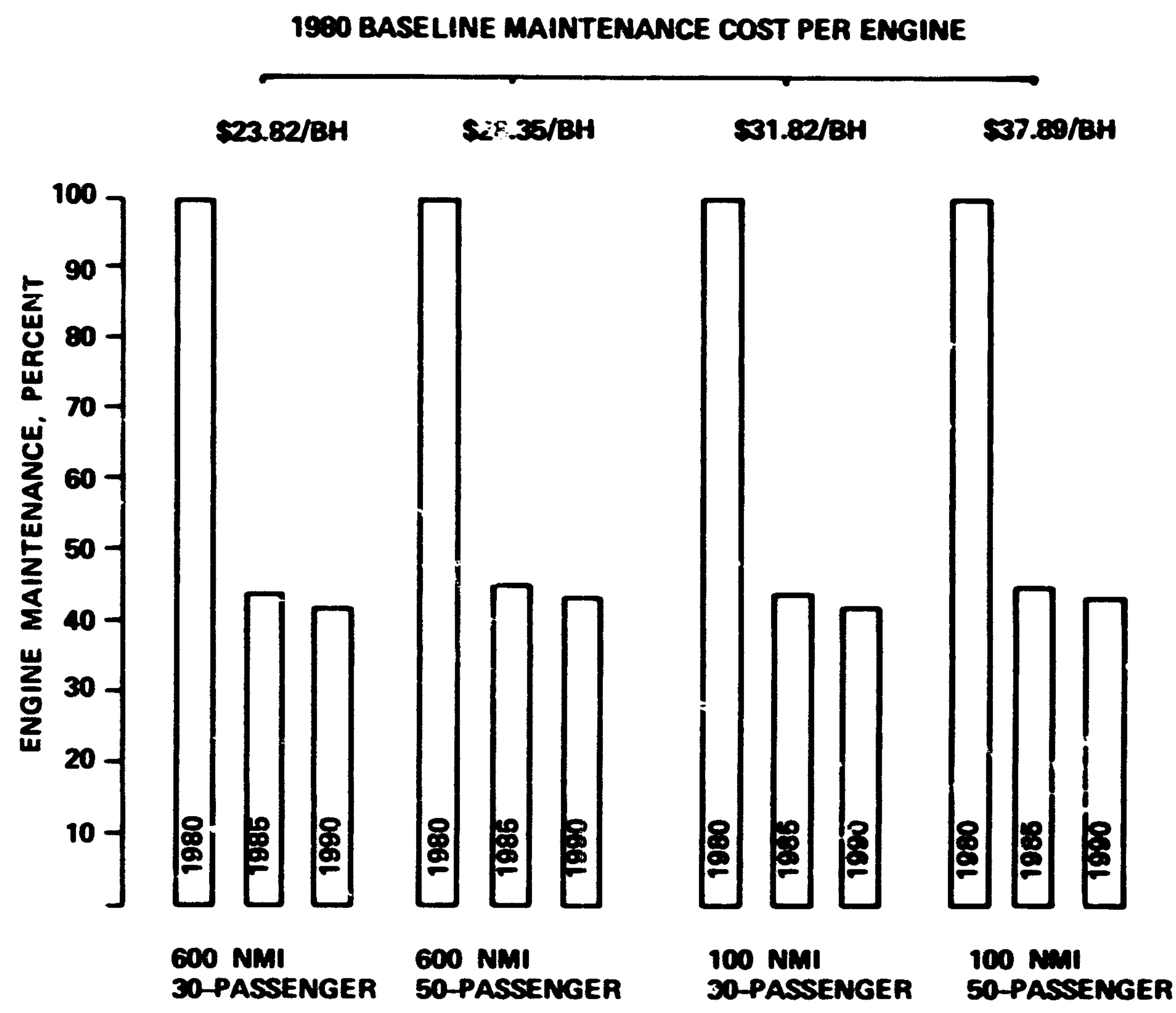


Figure 16. STAT Engine Maintenance Costs Relative to 1980 Baseline.

4.0 ADVANCED PROPULSION TECHNOLOGY AND DESIGN FEATURES

This section addresses the following elements of the study:

- o Identification and evaluation of alternate component technologies
- o Evaluation of candidate design features
- o Evaluation of alternate cycles and configurations
- o Propulsion evaluation

4.1 Identification and Evaluation of Alternate Component Technologies

The 1990 technology engine described earlier was the basis for more detailed engine trade-off studies oriented specifically towards alternate component technologies. In selecting this configuration, detailed trade-off studies were not performed. The engine, described in detail in a subsequent section does, however, incorporate a significant amount of advanced technology. A large number of alternate technologies were evaluated. All alternates could be available in the 1990 time frame assuming successful development. They do represent various levels of technical and development cost risk.

The evaluation of the alternate technologies was made with respect to the 1990 technology engine. Therefore the benefits/penalties are relative to a 1990 technology standard. No attempt was made to evaluate the inherent technology benefits in the 1990 engine relative to the 1980 technology engine, except at the overall engine level. The large difference in configuration between the 1990 and 1980 engine precludes a detailed comparison of this type.

4.1.1 Description of the 1990 Technology Engine

A cross section of the 1990 technology engine is shown in Figure 8.

4.1.1.1 Compressor

The NASA Small Axial/Centrifugal Design Study program showed that two-stage centrifugal compressors were competitive with axial/centrifugals and superior to single-stage centrifugals and multistage axials in the 10-lb/sec inlet corrected flow class. At 16:1 design pressure ratio, the two-stage centrifugal selected has an efficiency 3 points higher than current technology. This efficiency improvement is due to advanced 3-D blading and improved clearance control.

4.1.1.2 Combustor

The combustor is fabricated by rolling, stamping and welding INCO 617. Thermal-barrier coatings for improved durability are utilized, and the fuel injectors are the airblast atomizing type.

4.1.1.3 High-Pressure Turbine

The turbine is a two-stage cooled axial design. The inserted blades are directionally solidified (DS) MAR-M 247. The super-waspaloy disk is forged and machined. Advanced clearance control techniques (passive) and 3-D blading provide an efficiency that is approximately 4 points higher than current technology.

4.1.1.2 Low-Pressure Turbine

The low-pressure turbine is a two-stage, uncooled, axial design. The inserted solid DS MAR-M 247 blades have integral shrouds.

4.1.1.5 General

The speed reduction (propeller) gearbox is integral to the engine. A three-step reduction is utilized. Aircraft accessories mount on the cast gearbox housing.

A digital electronic fuel control with hydromechanical backup (and diagnostic and power management capabilities) was assumed.

4.1.2 Alternate Technology Evaluation

A large number of candidate advanced technology features were evaluated. The initial list of candidates was screened and those shown in Table XI were evaluated in more detail. The overall results of the alternate technology evaluation are shown in Table XII. Each technology area is discussed below.

4.1.2.1 Compressor

The 3-D blade shapes utilized in the two-stage centrifugal compressor entail expensive machining operations. Near net-shape powder metal (PM) fabrication techniques offer a potentially less expensive manufacturing method. Powder metal titanium and aluminum approaches were investigated. New PM aluminum alloys are under development for use up to temperatures of 650°F which would be appropriate for the first impeller. Use of aluminum rather than titanium offers a reduction of 40-percent weight and 60-percent cost. There is, however, a reduction of 30 percent in life. The impact on DOC for the PM aluminum first-stage impeller was approximately 1/2 of 1 percent. The PM titanium impeller does not offer a weight decrease but its life is equivalent to the

TABLE XI. STAT ALTERNATE ADVANCED TECHNOLOGY FEATURES

Compressor

- Single-Stage 10:1 or 12:1 Centrifugal
- Two-Stage 20:1 Centrifugal
- Powder Metallurgy (Alum and/or Ti)
- Axial-Centrifugal 20:1 (Single & Two Spool)

Combustor

- Machined Ring Fabrication
- Photo ETched Fabrication

Gearbox

- Laser Hardened Gears
- Composite Housing
- Super Plastic Forming/Diffusion Bonded (SPF/DB) Housing
- Traction Drive/Roller Gears

High-Pressure Turbine

- Cooled Single-Crystal Blades
- Uncooled
- Cooling Flow Modulation
- Single-Stage
- Active Clearance Control
- Tip Treatment
- Net-Shape Powder Metallurgy Disk
- 1500°F Disk Alloy

Low-Pressure Turbine

- Single-Stage
- Active Clearance Control
- Titanium Aluminide Second Stage

TABLE XII. ADVANCED TECHNOLOGY EVALUATION RESULTS

(100-nmi Mission)
$0.264/kl ($1.00/Gal) Fuel Cost

		Percent DOC Change		Percent Change in Block Fuel	
		30 Pax	50 Pax	30 Pax	50 Pax
	Compressor				
1	Powdered Aluminum First-Stage Centrifugal	-0.07	-0.05	-0.04	0
2	Powdered Titanium Second-Stage Centrifugal	-0.22	-0.12	0	0
3	Single-Stage 12:1 Centrifugal	+3.13	+2.87	+3.86	+3.47
4	Two-Stage 20:1 Centrifugal	-0.30	-0.40	-0.81	-0.74
5	20:1 Axial-Centrifugal (4 and 5 Axial Stages)	-0.88	-1.03	-2.44	-2.28
6	20:1 Two-Spool Axial-Centrifugal	-0.33	-0.62	-2.44	-2.27
	High-Pressure Turbine				
7	Single-Stage HPT	-1.21	-0.77	+0.38	+0.33
8	Uncooled [1477 K (2200°F), CRS, MA 6000, SC]	+1.06	+0.79	+0.44	+0.35
9	Cast Blades (SC, NASAIR-100)	+0.58	+0.11	-0.29	-0.32
10	Tip Treatment	-0.40	-0.42	-0.44	-0.46
11	Cooling Flow Modulation	-0.40	-0.42	-0.44	-0.46
12	Active Clearance Control	-0.43	-0.35	-0.48	-0.38
13	Net Shape PM Disk	+0.11	+0.03	0	0
14	1089 K (1500°F) Disk Alloy	-0.09	-0.07	0	0
	Low-Pressure Turbine				
15	Active Clearance Control	-0.58	-0.58	-0.64	-0.63
16	Titanium-Aluminide Second Stage	+0.33	+0.23	0	0
17	Single-Stage LPT	+0.65	+0.60	+1.27	+1.10
	Gearbox				
18	Laser Hardened Gears	-0.13	-0.11	0	0
19	Roller Gears	+0.51	+0.45	+0.05	+0.04
20	Composite Housing	+0.56	+0.39	+0.01	0
21	SPF/DB Gearbox	+3.80	+3.06	0	0
	Combustor				
22	Machined Ring Burner	+0.02	+0.02	0	0
23	Photo Etched Burner	-0.15	-0.12	0	0

machined titanium impeller. The reduction in cost relative to the machined titanium impeller is 40 percent. The net effect on DOC is a reduction of 0.1 to 0.2 percent. If a centrifugal compressor is used in the engine, two-stage centrifugal or axial/centrifugal, it is recommended that PM titanium impellers be selected.

Reference to Table XII shows that the most attractive compressor is a 20:1 pressure ratio axial/centrifugal. The single-stage centrifugal offers lower acquisition cost, lower maintenance cost and lower weight, but suffers a significant performance penalty. The 20:1 two-stage centrifugal and the two spool 20:1 axial/centrifugal both offer DOC benefits but do not offer as much as the 20:1 single-spool axial/centrifugal. The improvement in DOC is due primarily to an improvement in compressor efficiency (2 points higher than 20:1 centrifugal) and the effect of the higher cycle pressure ratio. Weight contributes only a small DOC benefit although the axial/centrifugal compressor is 25-percent lighter than the two-stage centrifugal. As evaluated, the axial/centrifugal compressor costs twice as much as the two-stage centrifugal and increases the maintenance cost of the engine by 15 percent. The increase in maintenance cost was due primarily to the higher cost of the component. If the cost penalty could be reduced by 50 percent, the DOC benefit could be doubled.

4.1.2.2 High-Pressure Turbine

The largest improvement in the high-pressure turbine was due to the substitution of a single-stage turbine for the two-stage design. It should be noted, however, that the trade-off study was performed for the engine utilizing a 16:1 pressure ratio, two-stage centrifugal compressor. A similar trade-off study for the 20:1 axial/centrifugal engine did not show a benefit.

The advantages of the single-stage turbine are:

- 8-percent reduction in engine weight
- 8.6-percent reduction in engine cost
- 6-percent reduction in maintenance cost
- 22-percent reduction in cooling flow

There is a significant performance penalty. The substantial increase in mean work coefficient of the single-stage design results in a 2 point loss in turbine efficiency.

Other significant technologies included tip treatment, cooling flow modulation and active clearance control. The primary factor in these technologies was the effect on fuel consumption, through changes in component efficiency and cooling flow. There were very small changes in weight, cost and maintenance but the effort required to quantify them was not warranted due to the relative insensitivity of DOC to small weight, cost and maintenance changes.

Advanced high strength directionally recrystallized PM and cast single-crystal (SC) nickel-base alloys that will allow uncooled turbine rotor inlet temperatures of 2200°F are currently in an early stage of development. A trade study was made to determine whether an uncooled turbine was more attractive than a cooled approach. Weight, cost and life of the uncooled turbine were all adversely affected. If life were held constant, turbine weight would increase significantly. The effect on performance was positive due to the elimination of blade and vane cooling but it was offset by a decrease in turbine efficiency of 1 point due to the high taper ratio design required.

The use of higher temperature capability materials in the turbine blades and vanes also had a disappointing result. The major reason for the negative results was the very high cost presently predicted for the higher temperature materials. If costs can be reduced, these materials would be more attractive.

A near net-shape PM disk (Rene 95) and a dual-alloy disk were investigated but offered little or no benefit. The disk environment is not severe enough to warrant use of these higher cost approaches. Little or no disk cooling or weight reductions are possible.

4.1.2.3 Low-Pressure Turbine

The low-pressure turbine initially proposed for the 1990 technology engine benefited only from the addition of active clearance control. Alternate materials such as titanium aluminide in the second stage, and a substitution of a single-stage design were not beneficial. The TiAl blades reduced the second-stage blade weight by 50 percent, but resulted in a higher cost and lower life.

4.1.2.4 Gearbox

Of the four alternate gearbox technologies investigated, only laser-hardened gears offered a positive benefit. A re-evaluation of the technology status of laser hardening resulted in the conclusion that it is not a 1990 technology item. Development of this technology is quite advanced, and it should be ready to transfer to production engines in a few years without large expenditures of R&D funds.

Roller gears and traction drive were investigated as an alternative to the conventional epicyclic gear train. Both approaches incur large weight penalties without offsetting cost, performance or maintenance advantages.

Composites and superplastic forming and diffusion bonding of titanium sheet (SPF/DB) were investigated for fabrication of the gearboxes. Both offered significant weight and maintenance advantages, but the composite gearbox was estimated to cost three times the cast aluminum gearbox and the SPF/DB gearbox was estimated to be 5 times more expensive.

4.1.2.5 Combustor

The two alternate combustor technologies investigated yielded small improvements. The machined ring combustor liner provides significant durability advantages (2x life) but at a 50-percent cost increase. The increased liner life is due to the elimination of hot spot due to sheet metal tolerances and the elimination of double thicknesses. Machined ring combustors could be selected for an advanced engine based on more detailed trade-off studies.

The photoetched combustor allows the substitution of coarse transpiration cooling for film cooling. Cooling passages and the large number of cooling air orifices are photoetched into the combustor liner. A photoetched burner would be 20-percent lower in cost and have twice the life of a conventional burner.

4.2 Design Features

In addition to the advanced-technology items discussed above, several engine design features were examined for their effect on the baseline airplane. The design features examined are listed in Table XIII. The selection of these features was based primarily on discussions with commuter operators, whose consensus was that engines designed for their use should be "as simple and maintainable as possible." Accordingly, since engine maintenance represents a significant portion of the commuters' operating cost, design features related to maintenance received the greatest emphasis in this portion of the study.

The design features that were specifically included or assumed in the design of the 1985 and 1990 engine were:

- o Modular design to permit on-the-wing replacement of major component groups such as the gearbox, compressor, HP turbine, and power turbine

- o Repair capabilities, such as extra balance material on high-speed rotors, threaded inserts to provide for replacement of pulled or loose studs, welded rather than

TABLE XIII. STAT DESIGN FEATURES

Maintenance	**Performance**
Modular Design	HP/LP Bleed
Repairability	Automatic Power Reserve
Consolidated Service and Check Points	
Standardized Tools/Tool Clearance	**Options**
	APU Operation
	QEC Nacelle
Diagnostic Systems	Starting Mode
Trend Monitoring	Flat Rating
Fault Isolation	Derating

brazed assemblies, inserted rather than integral blades, self-fixturing components to minimize tooling costs

- Consolidated service and checkpoints to minimize the number of inspection panels and reduce ground-check time; minimized lockwire requirements to permit checks and services by flight crew; provision of ample tool clearance and elimination of special tools and fixtures to conserve maintenance time and cost.

A separate analysis of each of the above features was not performed. These features, together with the durability design criteria resulted in the 55-percent reduction in maintenance cost mentioned earlier. The basis for the 1985 and 1990 engine maintenance cost was a detailed maintenance estimate using the Logistic Support Cost (LSC) computer model. Input to this model is a detailed estimate of the labor and material cost of all maintenance actions.

Other design features that were considered qualitatively but not included in the maintenance cost estimate were:

- Diagnostic systems that utilize engine-mounted sensors to calculate and record engine health parameters such as turbine temperature, power output, and SFC. Such systems can provide data for trend monitoring to allow routine servicing to be scheduled, as well as component data to allow isolation of individual component performance problems.

- Provision for adequate HP or LP bleed air to permit optimization of the engine cycle for the aircraft pneumatic systems.

- Automatic power reserve, in the event of a one-engine-inoperative (OEI) condition during takeoff to provide up to 10-percent additional power from the remaining engine.

- Engine operation as an auxiliary power unit (APU) by locking the propeller and operating the gas generator to provide bleed air. This would eliminate the need for either an on-board APU or a ground cart. However, this option has the disadvantages of high fuel consumption, high-temperature core exhaust-gas impingement on the stopped LP turbine, and noise.

- Quick engine change (QEC) nacelle to permit rapid removal and replacement of the complete propulsion system. This system, illustrated in Figure 17, incorporates large removable external panels to expose the engine. The engine and propeller are removed after disconnecting the fuel, electrical, engine-control system lines, and four mounting bolts located at Station A.

- Electric or pneumatic starting modes are both feasible for propulsion systems in the size class of this study. Pneumatic systems are almost universally employed by the larger carriers, while electric systems are prevalent in general aviation. Electric systems require on-board batteries, while pneumatic systems require either an on-board APU or a ground cart. The choice must be made by the individual commuter operator based on his experience and resources.

- Flat rating and derating are options that affect engine size, cost, and performance. While flat rating results in an engine that provides relatively high power at higher ambient temperatures, such an engine may actually be larger than necessary and would have relatively higher (and undesirable) SFC at the lower ambient temper atures in the flat-rating regime. It may be better to provide the smallest engine that will provide the required power with minimum margin. Derating is a form of providing margin for time-related performance deterioration. A derating level of 5 percent is typical; however, the percentage can vary depending on experience and anticipated operating conditions. The choice of these options and their levels is also an operator decision.

These features were addressed in a qualitative fashion only because it has been found that their benefit is highly dependent on a particular operator's usage. To arrive at meaningful quantitative results would require postulating route structures and an overall fleet mix.

4.3 Evaluation of Alternate Cycles and Configurations

Parametric analyses of various configurations, staging arrangements, and cycles applicable to an advanced engine were performed to determine what performance improvements over the 1990 baseline engine might be achieved. The basic engine types examined included a conventional free-turbine configuration and a single-shaft configuration. In addition, a low-spool driven compressor configuration was examined at discrete points. The mechanical arrangements of these configurations are illustrated

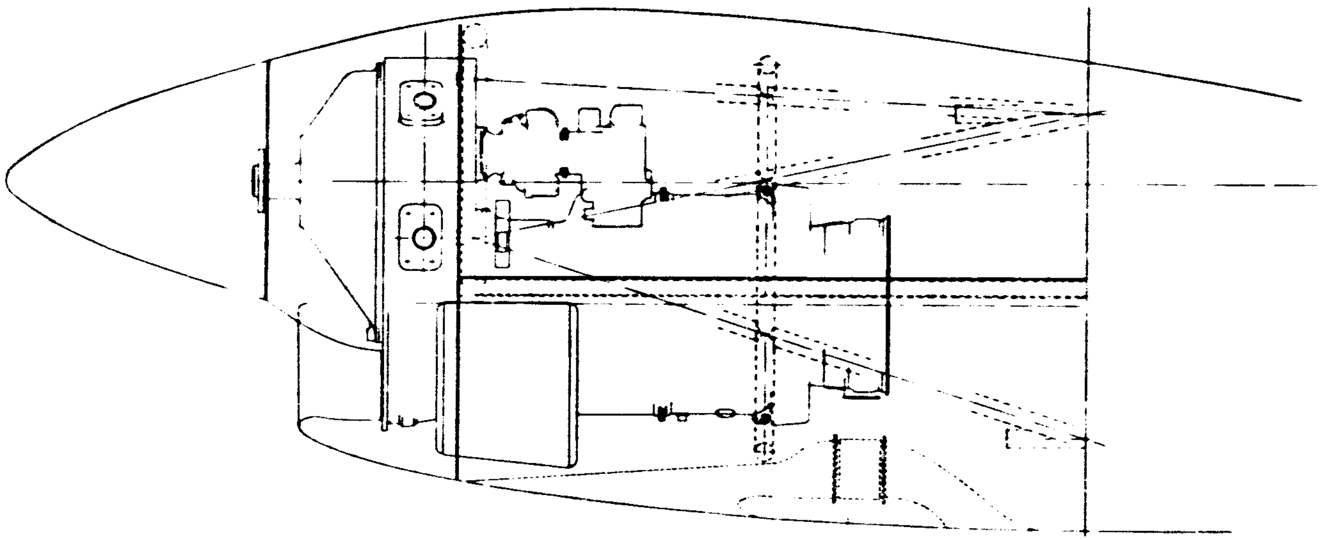


Figure 17. Quick-Engine-Change Nacelle.

schematically in Figure 18. The variation of component arrangements is indicated in Table XIV. For the free-turbine and single-shaft versions, design-point cycle analyses were performed for an OCR range from 10:1 to 20:1, and a TRIT range from 1477 K (2200°F) to 1700 K (2600°F). Compressor efficiencies were modified as a function of pressure ratio, as shown in Figure 19; turbine efficiencies were modified as a function of turbine work, as shown in Figure 20; and turbine cooling flows were scheduled as shown in Figure 21. Candidate alternate advanced technologies discussed in Section 4.2 are not included in the parametric evaluation with the specific exception of the single-stage turbine, single-stage centrifugal compressor and the 20:1 axial/centrifugal compressor.

The design point for these analyses was a cruise condition at 5180 m (17,000 ft), ISA. This condition, defined in Table XV, was chosen as a compromise between the STAT 100- and 600-nautical mile cruise conditions, and was retained throughout the study as the design condition for all subsequent performance comparisons.

The results of the parametric analyses are given in Figures 22 through 27 as curves of TSFC versus specific thrust (F/W). For these parameters, thrust includes propeller thrust (for which a propeller efficiency of 0.842 was assumed) and engine exhaust jet thrust. As noted previously, F/W is defined as the ratio of propeller plus jet thrust to the inlet airflow rate. These curves show performance only for the two-spool and single-shaft engines. A parametric analysis of this type was not performed for the low-spool drive configuration. A single design point calculation was performed for this configuration.

Figures 22 through 27 show, in general, that the minimum TSFC occurs at the highest pressure ratios for a given TRIT, and at a TRIT of approximately 1560 K (2350°F) for a given pressure ratio. Exceptions to this trend occurred with the engines that employed a single-stage centrifugal compressor, as shown in Figures 23 and 25. The minimum TSFC for these two engines occurred at pressure ratios near 14:1, since at higher pressure ratios the single-stage compressor efficiency (shown in Figure 19) falls well below those of the other compressor types.

A summary of the parametric analysis is shown in Table XVI. The first comparison shows engine at the cycle conditions chosen for the initial 1990 technology engine cycle, i.e., 16:1 pressure ratio and 2350°F. At these cycle conditions, the low-spool drive (G) and the axial centrifugal compressor configuration are superior to the two-stage centrifugal configuration which is the initial 1990 technology engine configuration. The TSFC improvement is in the order of 1 to 1.5 percent. The second comparison shows performance at the minimum TSFC cycle for each configuration. This comparison shows that the axial/centrifugal configuration (C) is 3-percent better in TSFC than the two-stage centrifugal configuration (A).

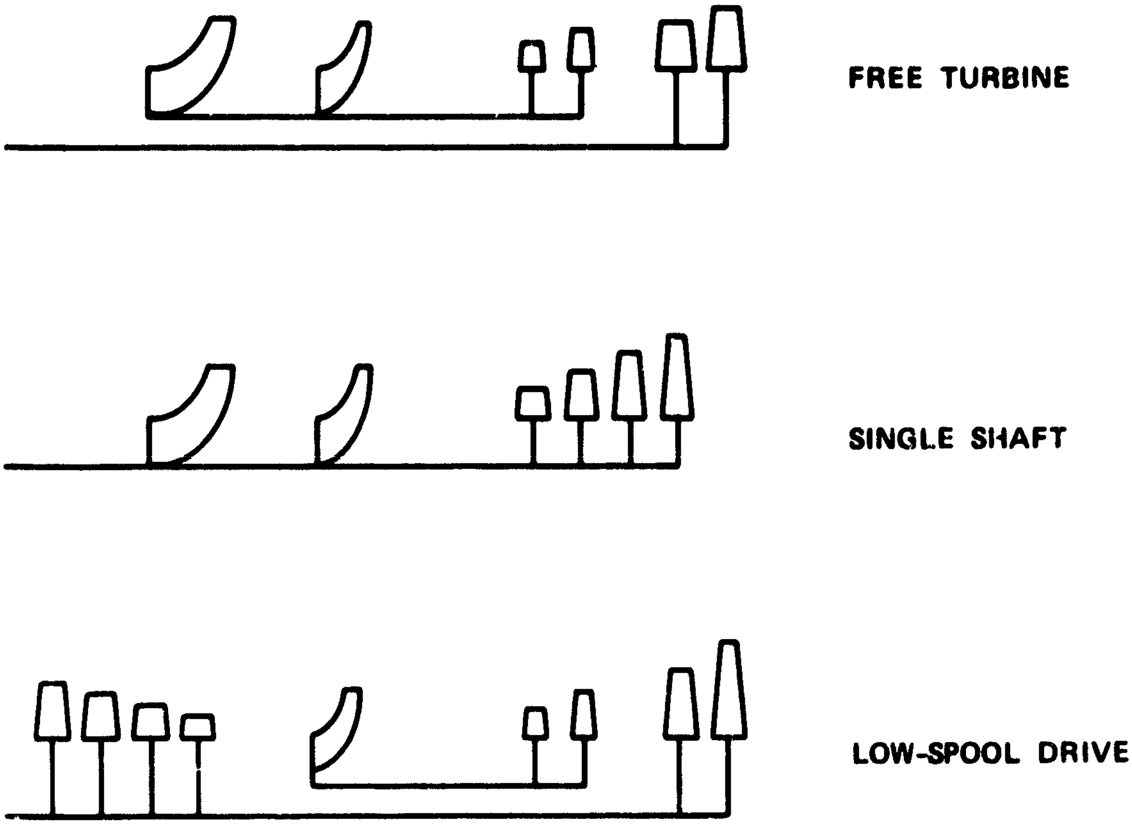


Figure 18. Parametric Study Basic Engine Configurations.

TABLE XIV. PARAMETRIC STUDY CONFIGURATIONS

	Compressor	Gas Generator Turbine	Power Turbine
Free Turbine			
A	2-Stage Centrifugal	2-Stage Axial	2-Stage Axial
B	1-Stage Centrifugal	2-Stage Axial	2-Stage Axial
C	Axial-Centrifugal	2-Stage Axial	2-Stage Axial
Single Shaft			
D	1-Stage Centrifugal	4-Stage Axial	---
E	2-Stage Centrifugal	4-Stage Axial	---
F	Axial-Centrifugal	4-Stage Axial	---
Low Spool Drive			
G	Axial/Centrifugal	2-Stage Axial	2-Stage Axial

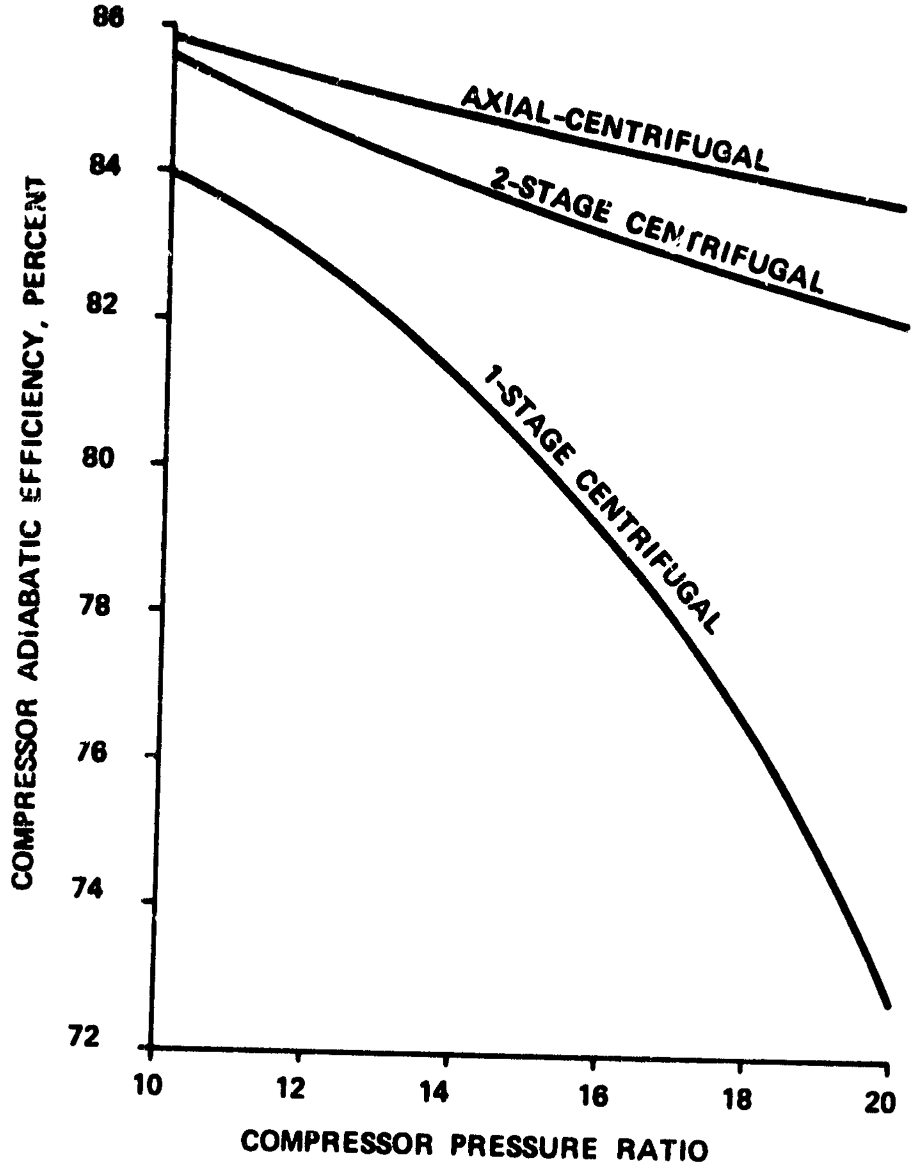


Figure 19. Compressor Efficiency Variation.

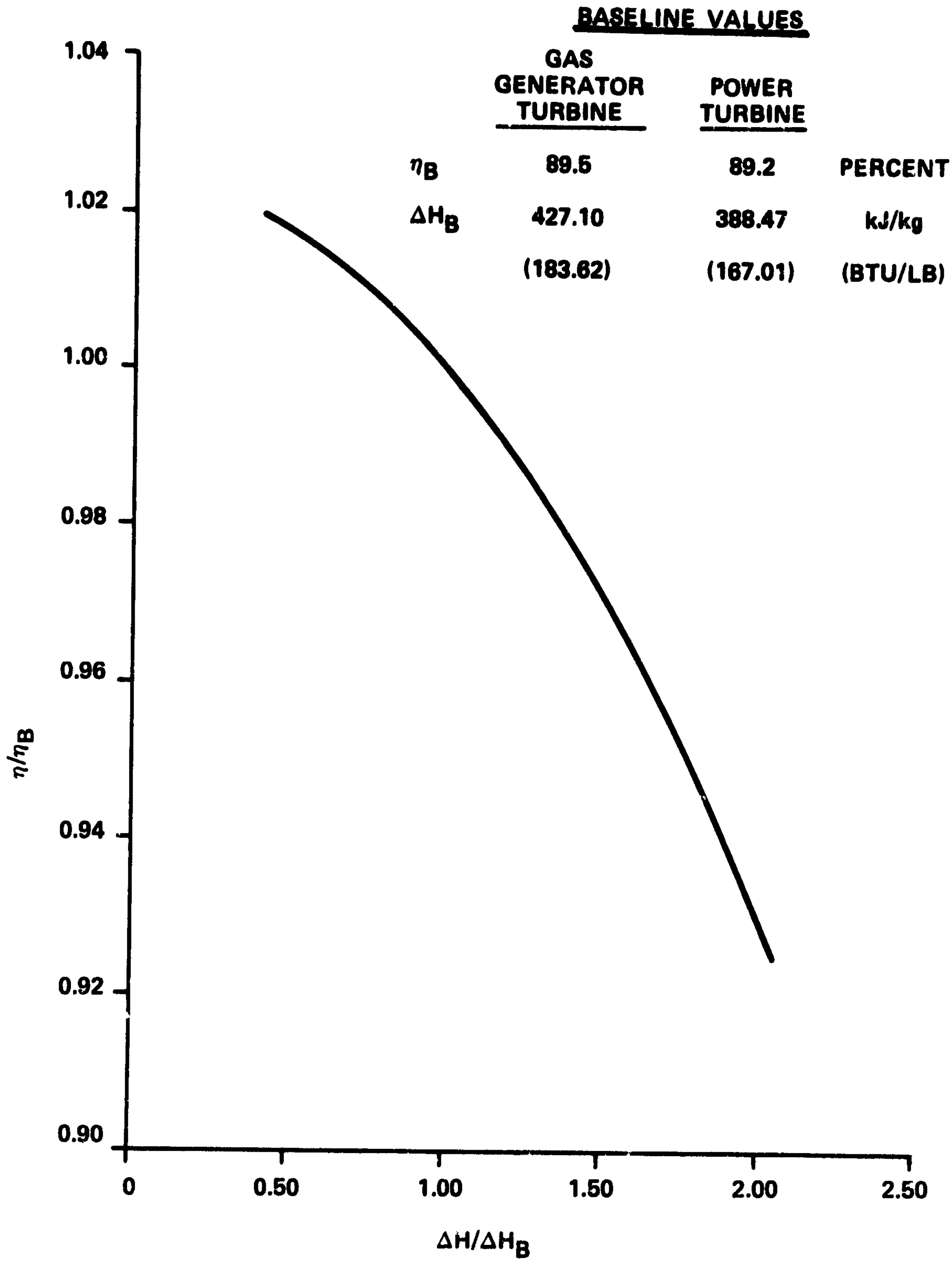


Figure 20. Turbine Efficiency Correlation.

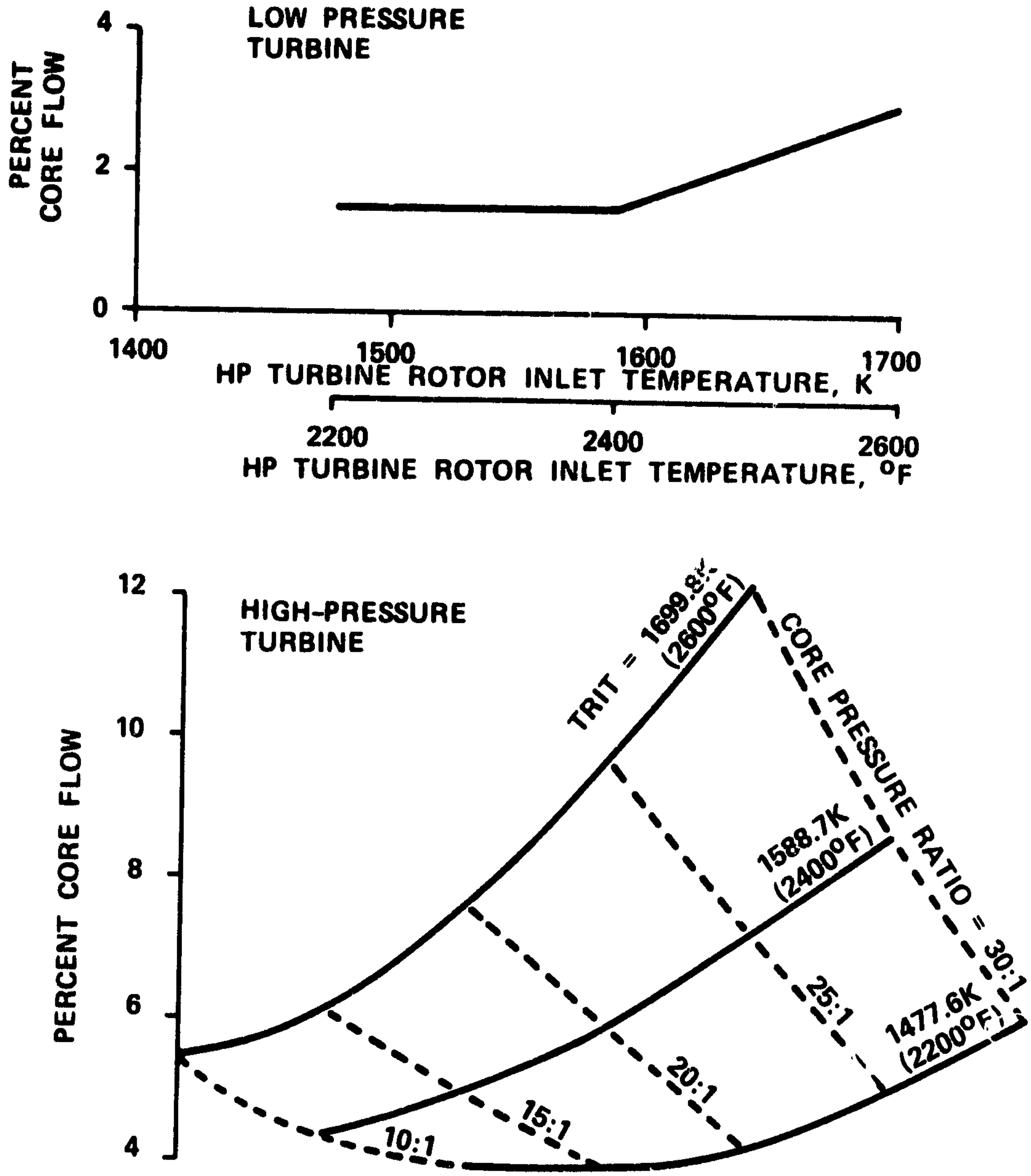
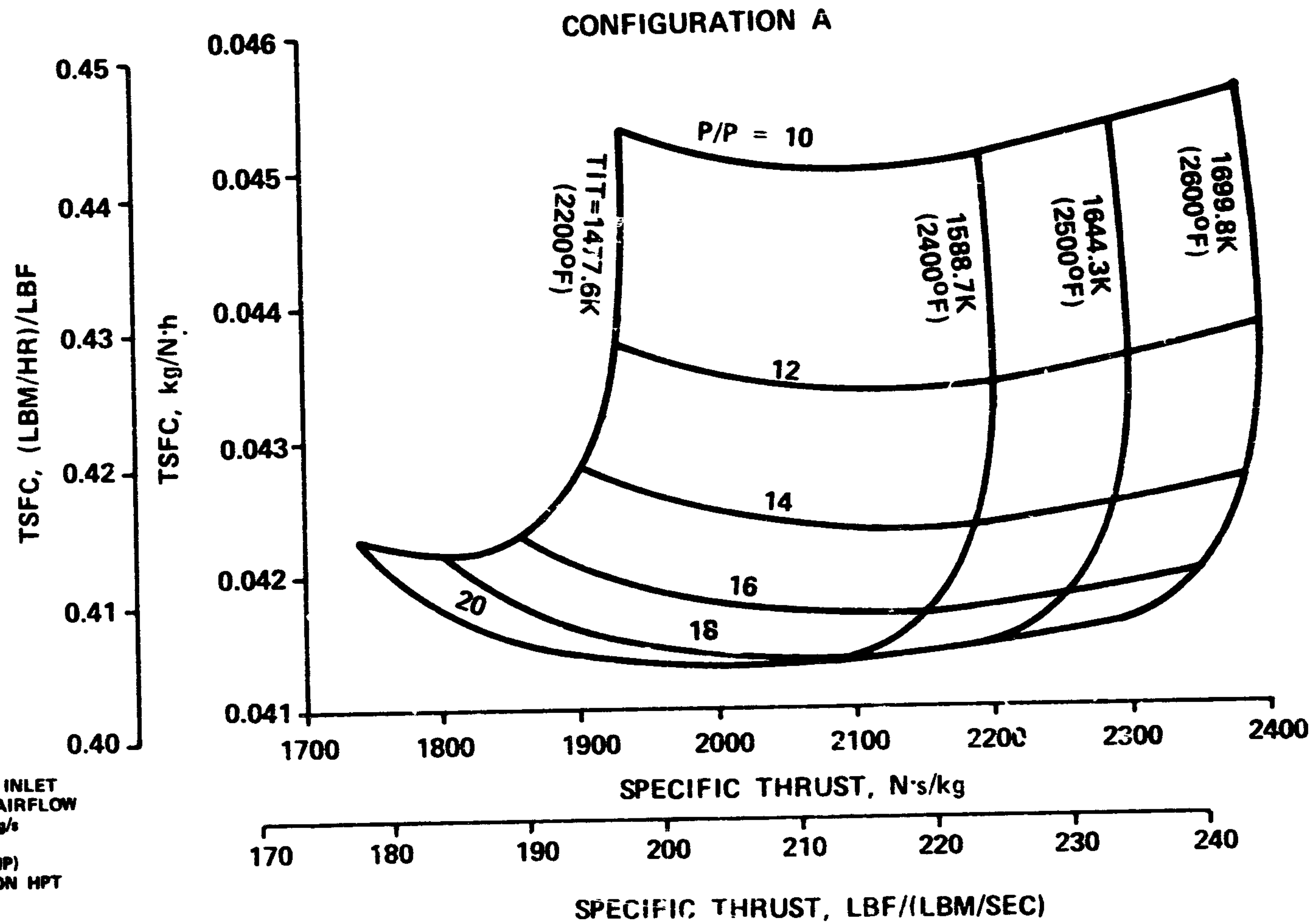


Figure 21. STAT Turbine Cooling Flow Schedules.

TABLE XV. PARAMETRIC ANALYSIS CONDITIONS.

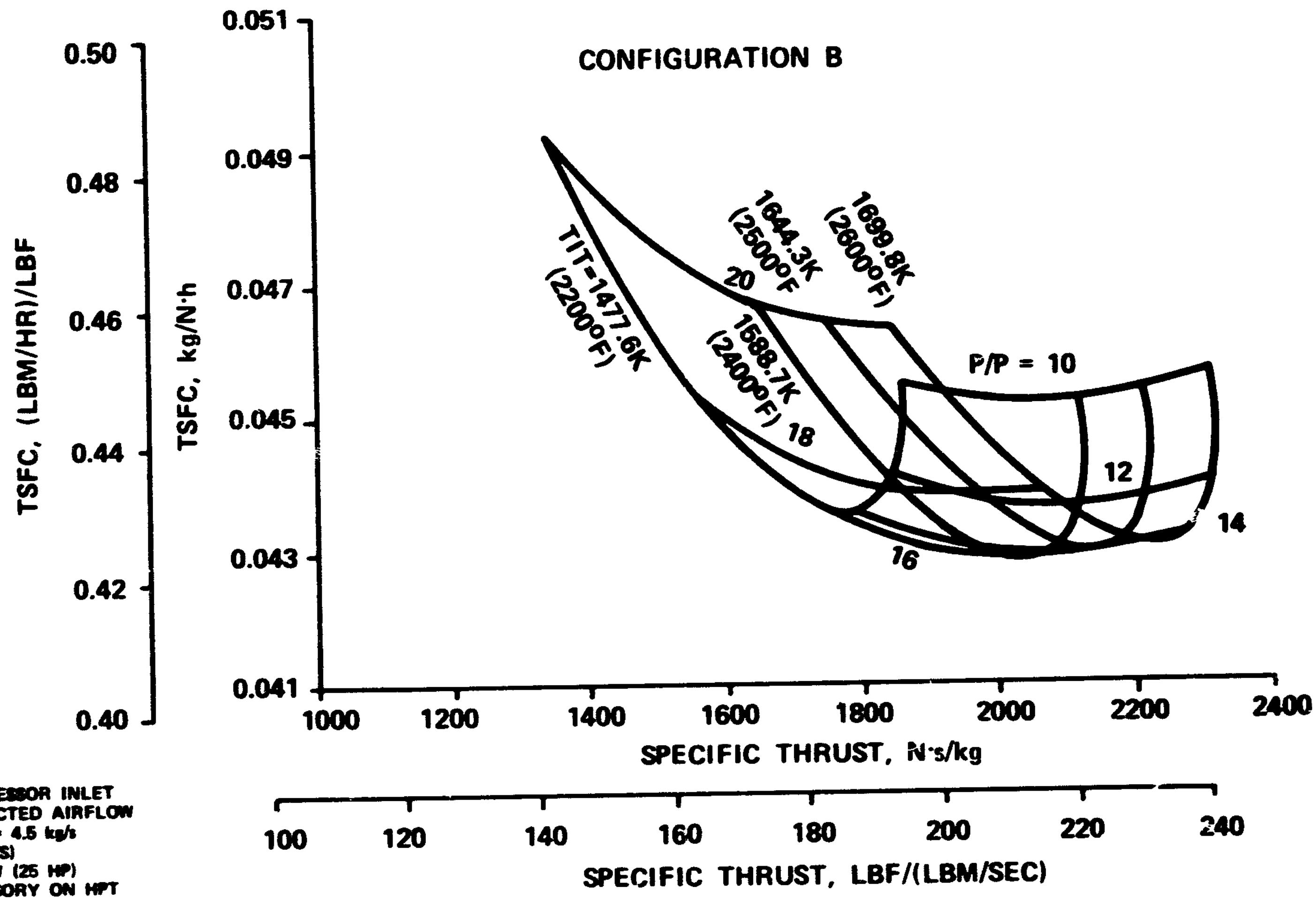
Altitude	5180 m (17,000 ft)
Ambient Temperature	255 K (-1.6°F)
Ambient Pressure	52.7 kPa (7.65 psia)
Flight Speed	291 KTAS (M=0.468)
Compressor Bleed Airflow	5.4 kg/min (12 lb/min)
Accessory Power	18.6 kW (25 hp)
Inlet Pressure Recovery	0.995



NOTES:

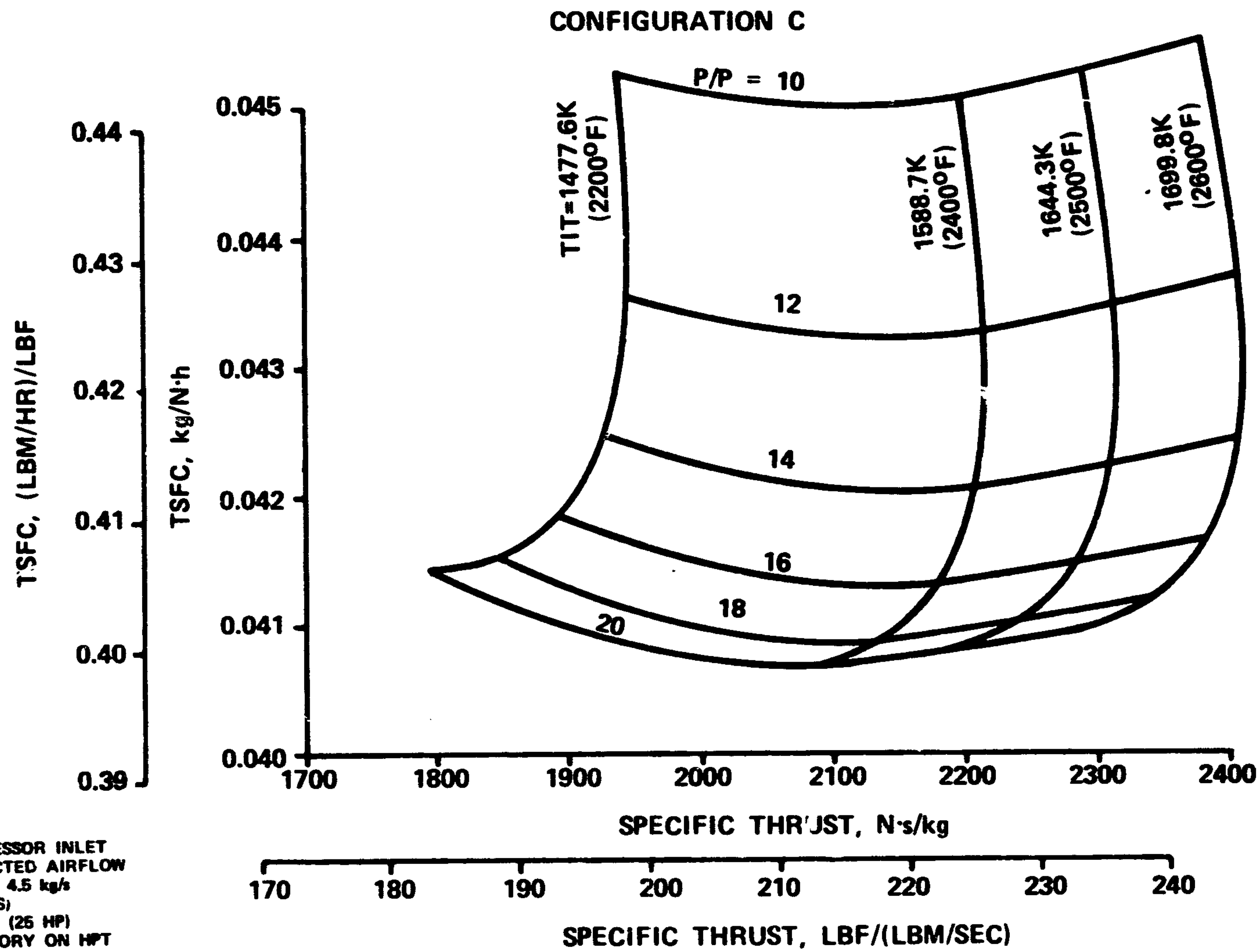
(1) COMPRESSOR INLET CORRECTED AIRFLOW RATE = 4.5 kg/s (10 LB/S)
(2) 18.6 kW (25 HP) ACCESSORY ON HPT
(3) 0.091 kg/s (12 LBM/MIN) COMPRESSOR BLEED
(4) ALTITUDE = 5181.6 m (17,000 FT)
(5) KTAS = 291
(6) INLET PRESSURE RECOVERY = 0.995
(7) WITHOUT ALTERNATIVE ADVANCED TECHNOLOGIES

Figure 22. STAT Two-Stage Centrifugal Compressor, Two-Stage HPT and LPT.



NOTES:
(1) COMPRESSOR INLET CORRECTED AIRFLOW RATE = 4.5 kg/s (10 LB/S)
(2) 18.6 kW (25 HP) ACCESSORY ON HPT
(3) 0.091 kg/s (12 LBM/MIN) COMPRESSOR BLEED
(4) ALTITUDE = 5181.6 m (17,000 FT)
(5) KTAS = 291
(6) INLET PRESSURE RECOVERY = 0.995
(7) WITHOUT ALTERNATIVE ADVANCED TECHNOLOGIES

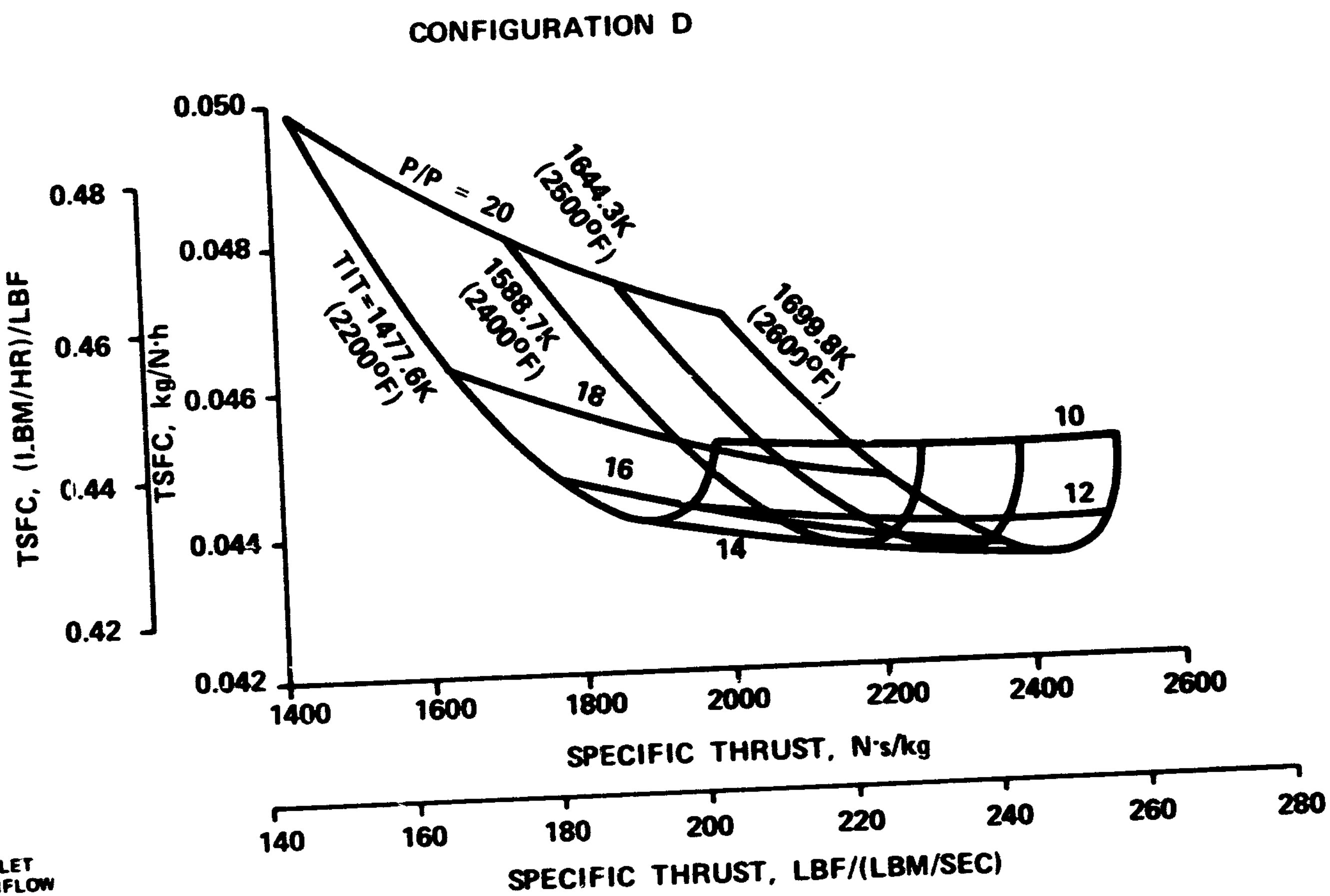
Figure 23. STAT Single-Stage Centrifugal Compressor, Two-Stage HPT and LPT.



NOTES:

(1) COMPRESSOR INLET CORRECTED AIRFLOW RATE = 4.5 kg/s (10 LB/S)
(2) 18.6 kW (25 HP) ACCESSORY ON HPT
(3) 0.091 kg/s (12 LBM/MIN) COMPRESSOR BLEED
(4) ALTITUDE = 5181.6 m (17,000 FT)
(5) KTAS = 291
(6) INLET PRESSURE RECOVERY = 0.995
(7) WITHOUT ALTERNATIVE ADVANCED TECHNOLOGIES

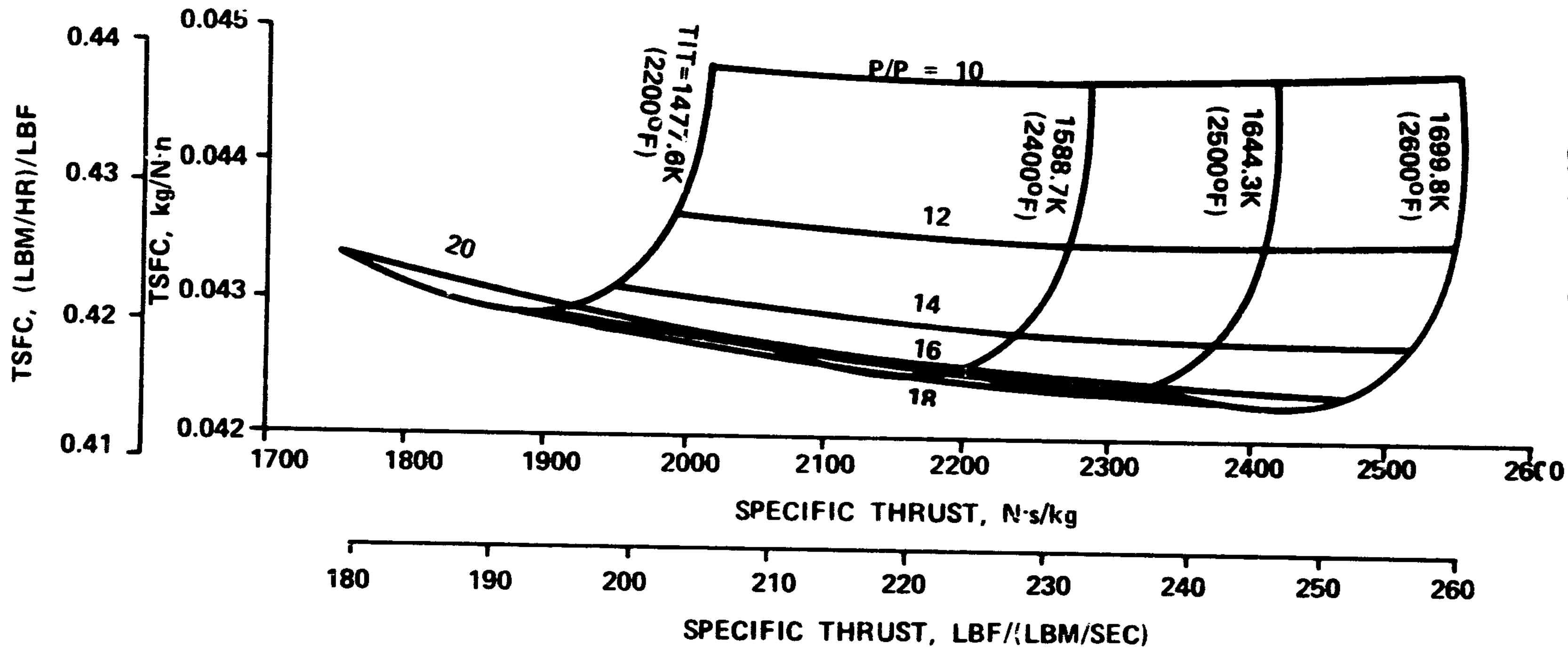
Figure 24. STAT Axial-Centrifugal Compressor, Two-Stage HPT and LPT.



NOTES:
(1) COMPRESSOR INLET CORRECTED AIRFLOW RATE = 4.5 kg/s (10 LB/S)
(2) 18.6 kW (25 HP) ACCESSORY ON HPT
(3) 0.091 kg/s (12 LBM/MIN) COMPRESSOR BLEED
(4) ALTITUDE = 5181.6 m (17,000 FT)
(5) KTAS = 291
(6) INLET PRESSURE RECOVERY = 0.995
(7) WITHOUT ALTERNATIVE ADVANCED TECHNOLOGIES

Figure 25. STAT Single-Stage Centrifugal Compressor, Single Shaft.

CONFIGURATION E

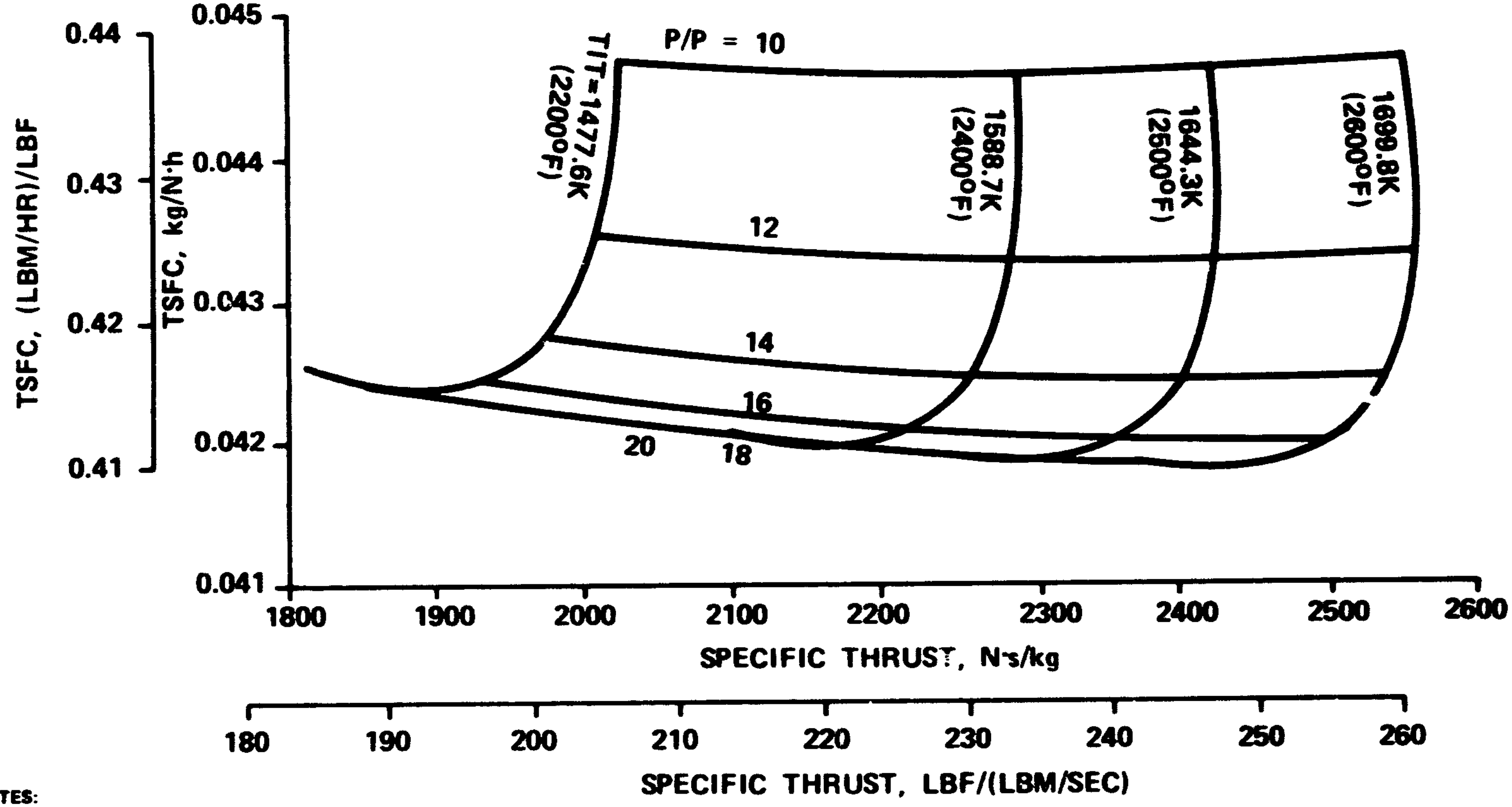


NOTES:

(1) COMPRESSOR INLET CORRECTED AIRFLOW RATE = 4.5 kg/s (10 LB/S)
(2) 18.6 kW (25 HP) ACCESSORY ON HPT
(3) 0.091 kg/s (12 LBM/MIN) COMPRESSOR BLEED
(4) ALTITUDE = 5181.6 m (17,000 FT)
(5) KTAS = 291
(6) INLET PRESSURE RECOVERY = 0.955
(7) WITHOUT ALTERNATIVE ADVANCED TECHNOLOGIES

Figure 26. STAT Two-Stage Centrifugal Compressor, Single Shaft.

CONFIGURATION F



NOTES:

(1) COMPRESSOR INLET CORRECTED AIRFLOW RATE = 4.5 kg/s (10 LB/S)
(2) 18.6 kW (25 HP) ACCESSORY ON HPT
(3) 0.091 kg/s (12 LBM/MIN) COMPRESSOR BLEED
(4) ALTITUDE = 5181.6 m (17,000 FT)
(5) KTAS = 291
(6) INLET PRESSURE RECOVERY = 0.995
(7) WITHOUT ALTERNATIVE ADVANCED TECHNOLOGIES

Figure 27. STAT Axial-Centrifugal Compressor, Single Shaft.

TABLE XVI. PARAMETRIC ANALYSIS PERFORMANCE COMPARISON

1990 Technology Engines (w/o Table XII Alternate Advanced Technologies)

Parameter	TRIT = 1560 K (2350°F) P/P = 16:1				TRIT and P/P Chosen For Minimum TSFC						
	F/W		TSFC		TRIT		P/P	F/W		TSFC	
	N-s/kg	lbf/ (lbm/sec)	kg/N-hr	(lbm/hr) /lbf	K	°F		N-s/kg	lbf/ (lmb/sec)	kg/N-hr	(lbm/hr) /lbf
Two-Spool:											
Configuration A	2089	213	0.0417	0.4090(3)	1589	2400	20:1	2030	207	0.0413	0.4054(2)
B	1961	200	0.0430	0.4220	1589	2400	15:1	2020	206	0.0430	0.4213
C	2099	214	0.0413	0.4051(2)	1589	2400	20:1	2079	212	0.0401	0.3937(1)
Single-Shaft:											
Configuration D	1971	201	0.0442	0.4335	1700	2600	14:1	2393	244	0.0431	0.4230
E	2118	216	0.0425	0.4172	1700	2600	17:1	2432	248	0.0422	0.4141
F	2137	218	0.0421	0.4131	1700	2600	19:1	2422	247	0.0418	0.4096(3)
Low-Spool Drive:											
Configuration G	2128	217	0.0411	0.4035(1)	-	-	-	-	-	-	-

NOTES:
(1) Compressor Inlet Corrected Airflow Rate = 4.5 Kg/s (10 lb/s)
(2) 18.6 kw (25 hp) HP Spool Horsepower Extraction
(3) 0.09 Kg/s (12 lb/min) Bleed Extraction (Compressor Discharge)
(4) 5181.6 m (17,000 ft) Pressure Altitude
(5) True Airspeed - 291 KTAS
(6) Inlet Pressure Recovery - 0.995
(7) Without Alternative Advanced Technologies

The performance benefit of the single low-spool drive engine analyzed was judged insufficient to warrant further investigation of this concept. The boosted or low-spool drive configuration is a viable configuration for fixed wing aircraft but has stability problems in a rotorcraft application. Adding compression stages to the low spool could be considered for growth engines applicable to fixed wing aircraft.

The primary result of the parametric investigations was to indicate a choice between the two-stage centrifugal and the axial/centrifugal compressor configurations and a pressure ratio of 16:1 and 20:1. This choice was also identified in the advanced technology evaluations where a complete evaluation of performance weight and cost was performed and showed a 1-percent DOC advantage for the axial/centrifugal 20:1 configuration relative to the two-stage centrifugal configuration at 16:1 pressure ratio. This result is tempered by the single-stage high pressure turbine trade-off study. This trade-off showed a 1 percent advantage for the single-stage high pressure turbine in the 16:1 two-stage centrifugal configuration. At the higher cycle pressure ratio, a two-stage turbine is required.

4.4 Propulsors

4.4.1 Propellers

4.4.1.1 Baseline Technology

Propulsors considered for the two airplanes defined in this study included only conventional propellers. Higher speed propellers and prop-fans, as defined by Hamilton Standard in data packages prepared for the STAT program, were not considered since the airplane cruise speeds selected by NASA-Ames were well within the speed regimes of conventional propellers.

The characteristics of the propellers, as defined by NASA-Ames for the 1980 baseline airplanes, are given in Table XVII, and the criteria for selection of the number of propeller blades are given in Figure 28. Evaluation of 3-, 4-, 5- and 6-bladed propellers resulted in minimum DOC with 5-bladed units for both the 30- and 50-passenger airplanes. Noise estimates were prepared for the FAR Part 36 locations and for the near field (fuselage surface). The noise levels were estimated with the methods given in the Hamilton Standard STAT propeller data packages.

The FAR Part 36, Stage 3, noise limits minus 8 EPNdB are also indicated in Table XVII. As noted previously (Table II), these levels were established as part of the baseline airplane performance requirements. The 30- and 50-passenger airplane noise levels were estimated by methods defined in the Hamilton Standard Red Book Supplement C, since the propeller design was based on

TABLE XVII. 1980 BASELINE AIRPLANE PROPELLER CHARACTERISTICS

Parameter	SI Units	Cust. Units	Airplane 30-PAX SI Value	Airplane 30-PAX Cust. Value	Airplane 50-PAX SI Value	Airplane 50-PAX Cust. Value
Number of Blades			5	5	5	5
Diameter	m	ft	3.86	12.65	4.84	15.89
Activity Factor			116.0	116.0	115.9	115.9
Design Integrated Lift Coefficient			0.50	0.50	0.50	0.50
Cruise Design Efficiency	%	%	88.5	88.5	88.5	88.5
Tip Speed	m/s	ft/sec	220.0	722.0	215.8	708.0
Weight, Each Propeller System*	kg	lb				
Noise: Far-Field (FAR Part 36, Stage 3-8 EPNdB)						
Takeoff (Limit: 89 - 8 = 81)		EPNdB		78.2		77.0
Sideline (Limit: 94 - 8 = 86)		EPNdB		80.4		79.6
Approach (Limit: 98 - 8 = 90)		EPNdB		78.0		76.7
Near-Field (Fuselage Surface)						
3658m (12,000 ft) cruise		dB		127.3		124.2
6096m (20,000 ft) cruise		dB		125.2		120.7
Acquisition Cost Per Propeller (1980 OEM)		$x100		163.0		251.3

*Gearbox weight is included with engine weight.

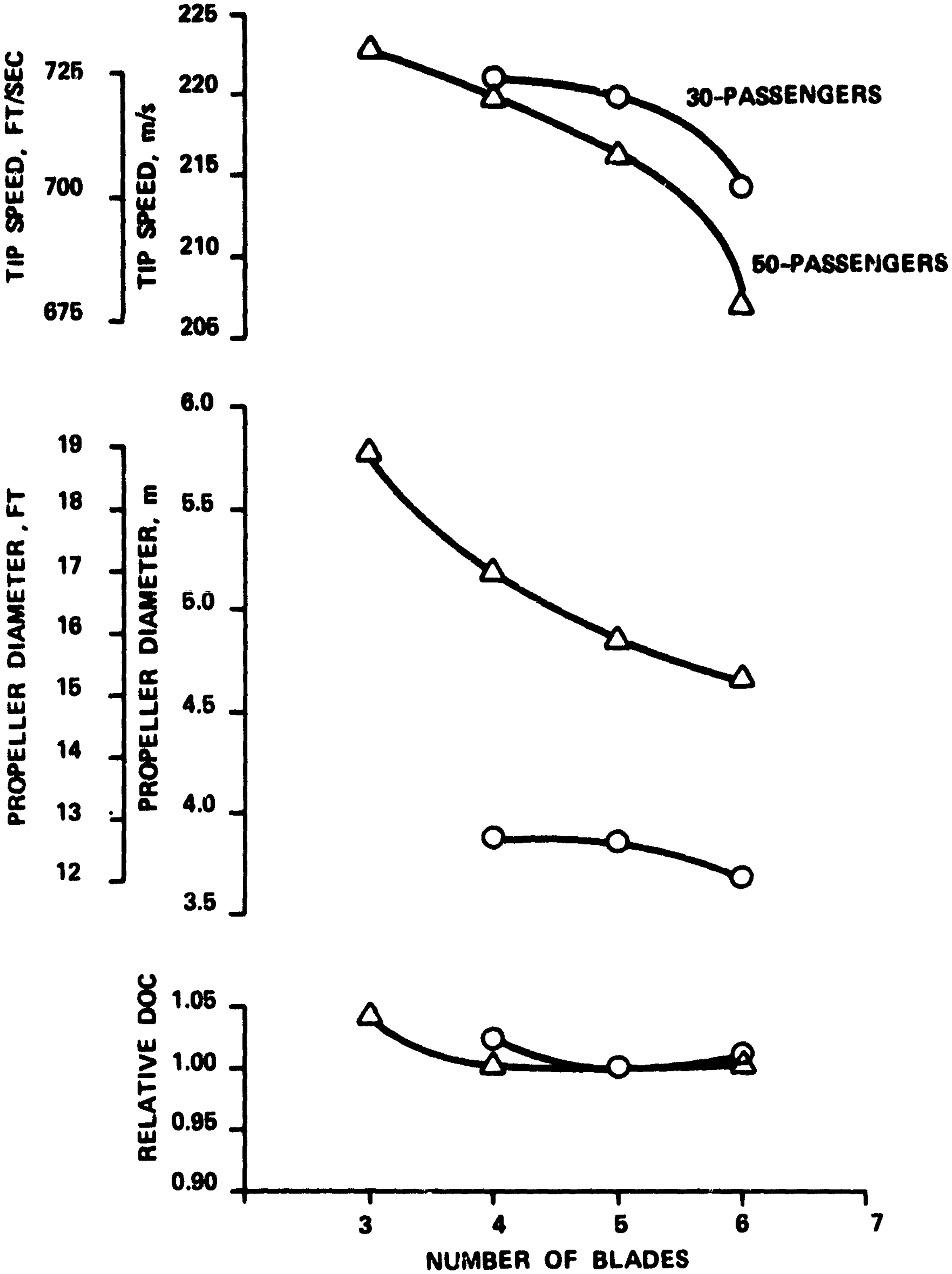


Figure 28. STAT Propeller Selection Criteria.

parameters given in that document. Corrections to EPNL were made according to the procedures given in the Advanced Technology Commuter Aircraft Propellers data package furnished by Hamilton Standard. The results are given in Table XVII.

The near-field noise levels at the fuselage surface show the same trend as the far-field levels in that the 50-passenger airplane levels are lower than those of the 30-passenger airplane. This is due to the greater diameter and the lower tip Mach number of the larger propeller. Correspondingly, the 50-passenger airplane required less acoustic treatment than the 30-passenger airplane [482 kg (1063 lb) versus 494 kg (1090 lb).

4.4.1.2 Improved Technology

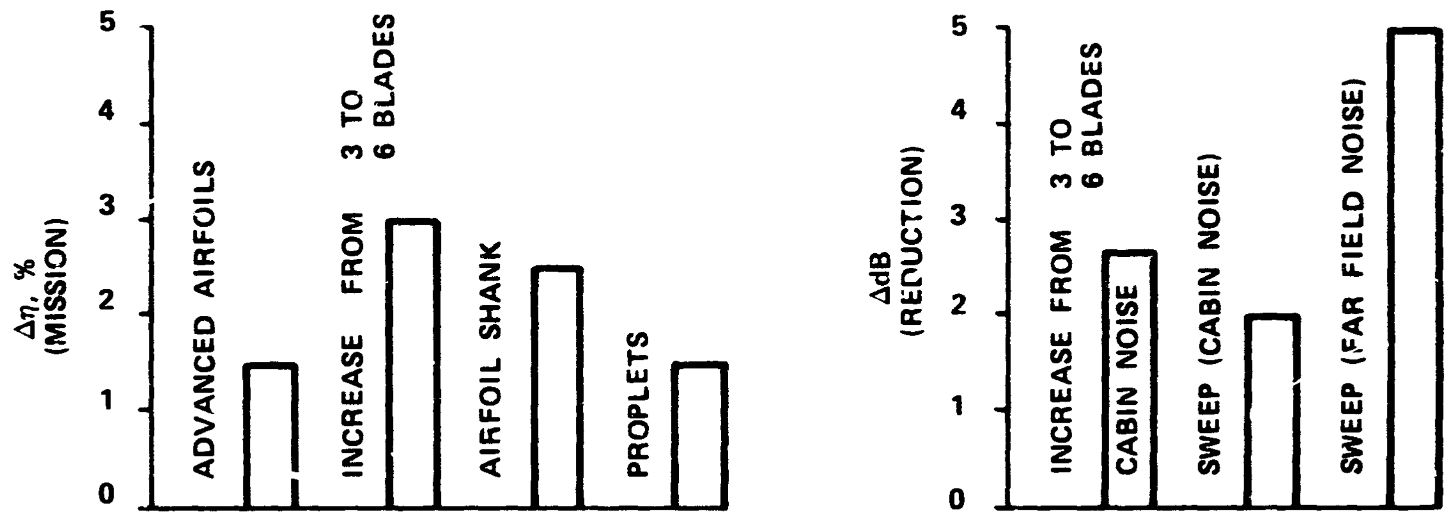
Numerous alternatives have been considered for improving propulsors. Some of these are listed in Table XVIII along with the areas in which improvements are expected. The improvements that result from these alternatives are summarized in Figure 29 for the 30-passenger airplane. These improvements are based on limited information and are subject to slight variation depending on specific propeller designs. Similar improvements would be expected for the 50-passenger, Mach=0.47 airplane. A more detailed breakdown of the effects of propeller efficiency, weight, and costs on the DOC for the 30- and 50-passenger airplanes is given in Table XIX. The parameters are given in relative values, with a current-technology propeller used as a baseline. Relative DOC values were determined from the airplane sensitivities furnished by NASA-Ames. As shown in Table XIX, a DOC reduction of approximately 4 percent is projected for the advanced STAT propeller configuration.

4.4.2 Comparison to Turbofan Engine

A comparison of the 1990 turboprop engines to a 1990 technology turbofan was conducted to verify that the turboprop cycle is the best solution for the STAT missions. The 1990 turbofan used in the comparison is a result of Garrett studies. It is a high bypass ratio, two-spool, concentric shaft design. A single-stage fan producing a pressure ratio of 1.55 is driven by a four-stage uncooled axial low-pressure turbine. The core compressor is an axial/centrifugal design which produces a pressure ratio of 15.3:1 and it is driven by a two-stage, cooled, axial turbine. The engine utilizes a reverse-flow annular combustor and mixed exhaust nozzle. Characteristics of the engine at the cruise condition of 3650 m (12,000 ft) and 0.452 Mach number are compared to the characteristics of the 30-passenger 1990 technology engine in Table XX. The turbofan was scaled to the thrust level of the turboprop engine assuming 84-percent propeller efficiency.

TABLE XVIII. PROPULSOR TECHNOLOGY IMPROVEMENT ALTERNATIVES

	Anticipated Improvement In		
	Performance	Noise	Weight
Airfoil Aerodynamics	x	x	
Shank/Spinner Integration	x		x
Spar/Shell Construction	x	x	x
Precision Synchrophasing		x	
Aero Elastic Tailoring	x		x
Blade Number Optimization	x	x	
Tip Sweep	x	x	
Proplets	x		
Counter-Rotation	x		



NOTE: $\Delta\eta$ ALSO INCLUDES EFFECTS OF OPTIMIZING DIAMETER AND TIP SPEEDS.

Figure 29. Projected Propeller Improvements.

TABLE XIX. ADVANCED PROPELLER EFFECTS ON DOC

Airplane	30-Passenger		50-Passenger	
Technology Level	Current	Advanced	Current	Advanced
Relative Efficiency	1.000	1.073	1.000	1.073
Relative DOC		0.954		0.945
ΔDOC, Percent		-4.6		-5.5
Relative Weight	1.000	1.204	1.000	1.205
Relative DOC		1.004		1.004
ΔDOC, Percent		+0.4		+0.4
Relative OEM Cost	1.000	2.554	1.000	2.357
Relative DOC		1.003		1.011
ΔDOC, Percent		+0.3		+1.1
Relative Maintenance Cost	1.000	1.615	1.000	1.500
Relative Doc		1.002		1.001
ΔDOC, Percent		+0.2		+0.1
Total DOC, Percent		-3.7		-3.[illegible]

NOTE: Relative DOC values based on sensitivities furnished by NASA-Ames.

TABLE XX. TURBOFAN AND TURBOPROP CHARACTERISTICS COMPARISON.

3650M (12,000 ft), 0.452 Mach No
Installed, Cruise Power 0.84 Propeller Efficiency

	Turbofan	30 PAX 1990 Turboprop
Net Propulsive Force, lb	1479	1479
Thrust Specific Fuel Consumption, lb/hr/lb	0.56	0.421
Fan Pressure Ratio	1.55	--
Bypass Ratio	6.67	--
Core Pressure Ratio	15.4	15.2
Overall Pressure Ratio	23.5	15.2
Turbine Inlet Temperature, °F	2350	2350
Core Corrected Flow, lb/sec	10.22	9.65
Weight, lb	546	623

The difference in fuel consumption between the turboprop and the turbofan is 33 percent. This large difference in fuel consumption would be offset to a degree by potentially lower airplane drag, lighter engine system weight and lower airframe acoustic weight. Engine acquisition and maintenance cost are expected to be near equal. The high sensitivity of DOC to fuel consumption and the relatively low sensitivity to weight, cost, and maintenance suggest that the turbofan is not competitive with the turboprop for short, low speed missions. At higher speeds and altitudes and for longer missions, this comparison would warrant more detailed investigation.

4.5 Final STAT Engine Selection

The major choice to be made in selecting the final advanced engine configuration is the type of compressor and the design pressure ratio. A 1-percent improvement is possible if the 20:1 axial/centrifugal compressor is selected. On the other hand, if the two-stage centrifugal compressor is retained, a 1-percent improvement is possible if a single-stage turbine is selected. In the first case, the DOC benefits from reduced fuel consumption and in the second case, the DOC benefit is due to reduced acquisition and maintenance cost and lower weight. It is recommended that both options should be pursued in research and technology programs.

A component-by-component description of all advanced technology features selected for the final 1990 engine configuration is given below.

4.5.1 Compresor Section

Powder metallurgy (PM), near-set-shape, titanium centrifugal compressor impellers are recommended for both stages of the two-stage centrifugal and the centrifugal portion of the axial/centrifugal compressor if that option is eventually selected.

The powdered aluminum first-stage impeller was not selected due to its lower durability.

For the final engine definition the two-stage centrifugal compressor will be retained to facilitate performance and configuration definition. However, since the differences between the two-stage centrifugal and axial/centrifugal configurations is relatively small, both options should be developed.

4.5.2 Turbine Section

The turbine selection will ultimately be determined by the compressor selection. A 16:1 pressure ratio design could use the single-stage HP turbine but the 20:1 pressure ratio design would require the two-stage HP turbine. Tip treatment, active clearance control, and cooling flow modulation are recommended for the final configuration.

4.5.3 Combustor Section

Advanced combustor fabrication methods - machined ring or photoetching - offer higher durability combustors. A combined program addressing advanced fabrication methods, improved cooling and thermal-barrier coatings would be a worthwhile research technology objective.

4.5.4 Gearbox

Laser hardened gears were selected for the final version of the engine. This technology is now in development and government sponsorship of additional programs is not considered necessary.

4.5.5 Combined Technologies

The combined effect of these changes is summarized in Table XXI. Three alternatives are shown. They are:

- Two-stage centrifugal with two-stage HP turbine
- Two-stage centrifugal with one-stage HP turbine
- Axial/centrifugal with two-stage HP turbine.

TABLE XXI. FINAL DOC BENEFITS 1990 ENGINE 100-NMI MISSION.

Compressor (P/P) HP Turbine (TIT)*	Configuration A Two-Stage Centrifugal (16:1) Two-Stage (2350°F)		Two-Stage Centrifugal (16:1) One-Stage (2350°F)		Configuration C Axial Centrifugal (20:1) Two-Stage (2350°F)	
	% Change DOC 30 PAX	DOX 50 PAX	% Change DOC 30 PAX	DOC 50 PAX	% Change DOC 30 PAX	DOC 50 PAX
Base*	-16.70	-15.40	-16.70	-15.40	-16.70	-15.40
Compressor Type	0	0	0	0	- .88	- 1.03
Turbine Type	0	0	- 1.21	- 0.77	0	0
PM Impeller(s)	- 0.44	- 0.24	- 0.44	- 0.24	- 0.22	- 0.12
HPT Tip Treatment	- 0.40	- 0.42	- 0.40	- 0.42	- 0.40	- 0.42
HPT Active Clearance Control	- 0.43	- 0.35	- 0.43	- 0.35	- 0.43	- 0.35
Cooling Flow Mod	- 0.40	- 0.42	- 0.40	- 0.42	- 0.40	- 0.42
LPT Act Clearance Control	- 0.58	- 0.58	- 0.58	- 0.58	- 0.58	- 0.58
Laser Hardened Gears	- 0.13	- 0.11	- 0.13	- 0.11	- 0.13	- 0.11
Photoetched Combustor	- 0.15	- 0.12	- 0.15	- 0.12	- 0.15	- 0.12
TOTAL**	-19.23	-17.64	-20.44	-18.41	-19.89	-18.55

*Cruise
**Relative to 1980 Engine

The overall benefit of advanced technology to the STAT aircraft is approximately 20-percent reduction in DOC for the 30-passenger design. Almost 60 percent of this improvement is due to component efficiency and cycle quality improvements. Most of the remaining 40-percent reduction is due to improvements in maintenance. Weight and acquisition cost did not have a large influence.

The difference in DOC improvement between the three configurations shown in Table XXI is approximately 1 percent. This difference does not warrant a definitive choice considering the level of engine designs allowed within the scope of this study. As stated previously, both axial/centrifugal and centrifugal compressor technology programs should be pursued. Technology programs addressing one- and two-stage turbine designs should also be implemented.

Appendix III provides detailed cycle and configuration data and off-design performance for the two-stage centrifugal (16:1 P/P), two-stage turbine configuration. Specific fuel consumption would be approximately 1.5-percent and 4.0-percent lower for the two-stage centrifugal/one-stage turbine and the axial/centrifugal/two-stage turbine configurations, respectively.

4.6 Benefit Assessment

An assessment of the benefits of the 1985 derivative engine and the 1990 advanced engine was performed. For the benefit assessment, the 1990 engine utilizing the two-stage centrifugal compressor and the two-stage high-pressure turbine was used. Factors considered were the engine SFC, weight, acquisition cost, maintenance cost, the impact of the advanced technologies, and the expected advantages of advanced propellers. The results of this assessment are given in Table XXII for the 100-nmi mission. (Since all economic evaluations were performed only for the 100-nmi mission, a similar assessment was not performed for the 600-nmi mission.) These results show that for the 1985 derivative engine, a DOC reduction of approximately 11 percent would be achieved without the benefits of advanced technologies or advanced propellers. It was not considered that either the advanced technologies or the advanced propellers would be sufficiently developed for incorporation in the 1985 derivative engine.

However, for the 1990 advanced engine, a 21- to 23-percent reduction in DOC is projected based on the use of the 1990 baseline engine with improved components, advanced technologies and the advanced propeller effects given in Table XIX. These benefits are dependent upon appropriate research and technology efforts as recommended in subsequent sections.

The final results of the benefit assessment are summarized in Table XXIII. This table shows absolute values of DOC, block fuel,

TABLE XXII. BENEFITS FOR 100-NMI MISSION*

Engine	1985 Derivative		1990 Advanced	
Passengers	30	50	30	50
ΔPercent DOC For:				
SFC	-4.5	-4.2	-9.3	-8.5
Weight	-0.6	-1.0	-0.7	-1.1
Acquisition Cost	+0.2	+0.1	+0.2	+0.1
Maintenance Cost	-6.7	-5.7	-6.9	-5.9
Advanced Technologies	---	---	-2.5	-2.2
Advanced Propellers	---	---	-3.7	-3.9
Total ΔDOC, Percent	-11.6	-10.8	-22.9	-21.5

*Fuel Cost at $0.264/ℓ ($1.00/gal)

TABLE XXIII. STAT TOTAL BENEFITS.

Engine Technology					Current				Derivative				Advanced			
Airplane Passengers					30		50		30		50		30		50	
			SI Unit	Cust Unit	SI Value	Cust Value	SI Value	Cust Value	SI Value	Cust Value	SI Value	Cust Value	SI Value	Cust Value	SI Value	Cust Value
Direct Operating Cost (DOC), $/BH																
Mission	Fuel $/l	Fuel $/Gal														
600 NMI	0.264	1.00				506.32		697.53		459.87		639.69		390.37		547.56
600 NMI	0.396	1.50				598.76		821.40		543.22		751.01		461.64		644.80
100 NMI	0.264	1.00				531.43		733.68		474.81		622.69		409.73		575.94
100 NMI	0.396	1.50				628.36		866.04		562.20		781.35		484.46		679.84
Mission Block Fuel																
600 NMI			kg	lb	1237.4	2728	1666.0	3673	1115.8	2460	1497.3	3301	938.1	2068	1250.6	2757
100 NMI			kg	lb	307.5	678	416.4	918	274.4	605	373.3	823	226.8	500	309.8	683
Engine Acquisition Cost				$x1000		230.0		308.9		227.6		306.2		224.8		303.1
Airplane Acquisition Cost				$x1000		3437.7		4804.8		3408.7		4769.1		3375.3		4728.0
5-Year Ownership Cost*																
Direct				$x1000		6595.5		9075.2		5903.4		8193.1		5085.1		7124.0
Indirect				$x1000		5312.4		7332.2		4758.7		6626.4		4277.3		6199.4
Total				$x1000		11,907.4		16,407.4		10,662.1		14,819.4		9362.4		13,243.4

*For Mixed Mission and
$0.264/ ($1.00/gal) Fuel Cost

NOTE: All costs in Constant 1980 dollars.

engine and airplane acquisition costs, and 5-year ownership costs for the three engine-technology levels and for the 30- and 50-passenger airplanes. In addition, values of DOC which correspond to those shown in Figures 9 through 14, are given for fuel costs of $0.264/l ($1.00/gal) and $0.396/l ($1.50/gal). As noted previously, the DOC's were calculated with the equation furnished by NASA-Ames but using Garrett engine-maintenance costs.

The 5-year ownership costs were estimated from a mission mix that was developed from a variety of data included in References 1 through 6. This mission mix is based on commuter fleet projections, typical commuter city-pair mileages, and commuter passenger mileage records. The mix assumes that during the decade following introduction of the STAT airplanes, the percentage of flights for various stage lengths will change for the 30- and 50-passenger airplanes; and that, at the end of the decade, a higher percentage of the larger airplanes will assume the longer stage lengths. This mission mix is illustrated in Figure 30, which shows the assumed percentages of flights by 30- and 50-passenger airplanes over various stage lengths for the current, derivative, and advanced-technology level STAT airplanes.

4.6.1 Alternative Applications

The application of STAT engine technology to areas other than commuter aircraft was also examined as a matter of course. Both military and commercial options were considered. Future military applications include rotary- and fixed-wing aircraft as well as automotive applications such as wheeled and tracked vehicles. The commercial market includes a broad mixture of aircraft, ground transportation, and other applications. These are discussed briefly below.

The core of the STAT engine is well suited as the basis for military turboprop, turbofan, or turboshaft applications. Turboprop- or turbofan-powered applications might include small-to-medium size transport airplanes for use as intracontinental cargo, passenger, or command carriers. Such transports could relieve larger aircraft for the larger, higher-volume missions. Turboshaft-powered applications might include medium-to-large helicopters [4,500 to 13,600 kg (10,000 to 30,000 lb) gross weight] designed as cargo or personnel carriers. Such helicopters could include advanced rotorcraft such as X-wing or tilt-rotors for the high-speed transport of small strike forces and their support equipment.

The numerous potential commercial markets that might utilize the STAT core and component technology include turbofan-, turboprop-, and turboshaft-engine applications. The improved cycle, the advanced technologies, and the design improvements are

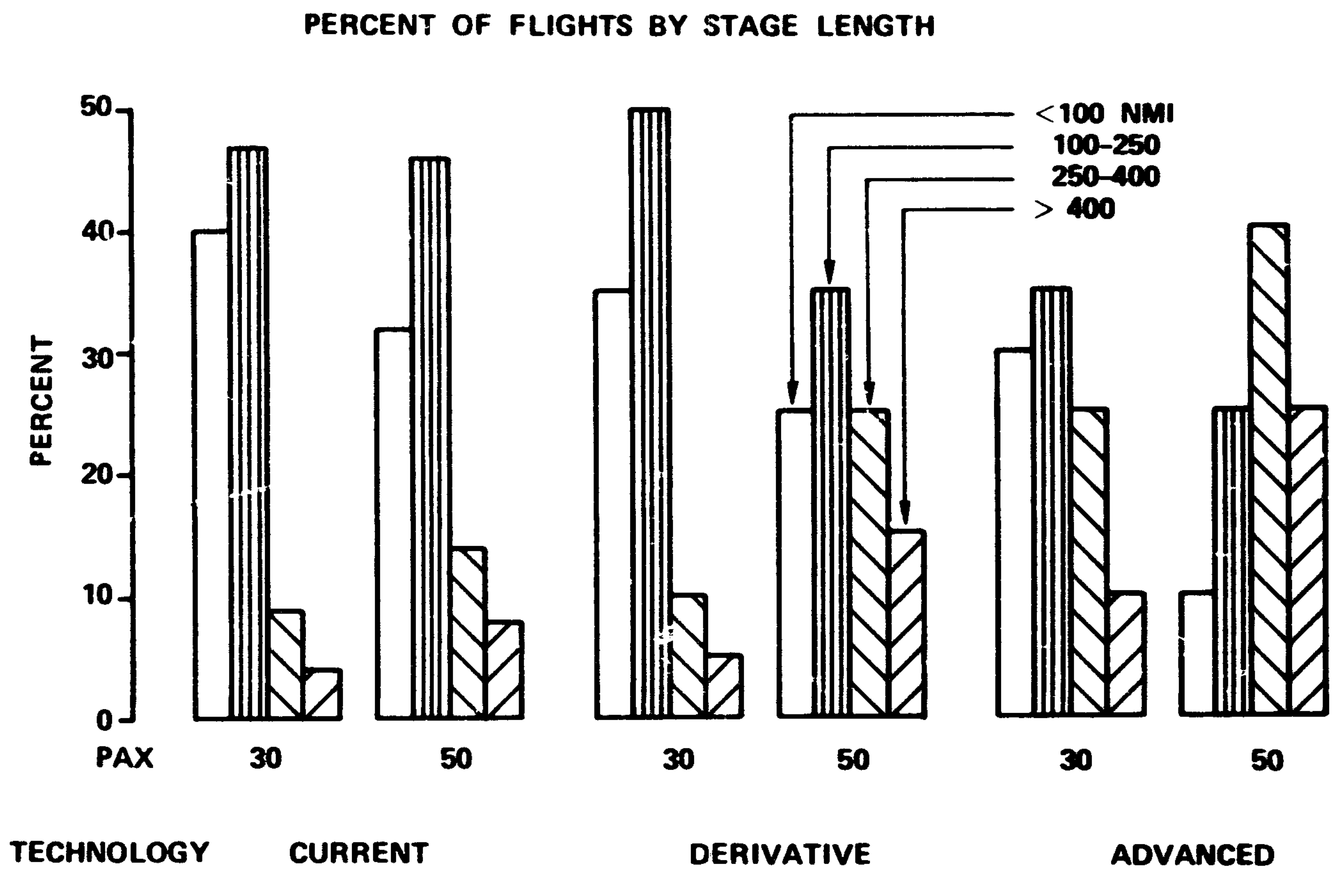


Figure 30. STAT Mission Mix.

features that may receive varied emphasis in the commercial environment; however, minimum fuel and maintenance costs are requirements in the competitive commercial community. Specific applications could include:

- General Aviation - Business and executive airplanes.
- Rotorcraft - Conventional cargo and passenger helicopter in the 4,500 to 13,600 kg (10,000-lb to 30,000-lb) gross weight class, single- and twin-engines. New higher-speed rotorcraft including tilt-rotors, compound helicopters, and the advancing-blade concept.
- Marine - Crew boats, hydrofoil passenger ferrys.
- Stationary Plants - Continuous-duty power plants for pumping or electrical generation; standby emergency plants for hospitals or similar critical requirements.

5.0 RECOMMENDED FUTURE RESEARCH

5.1 Program Scope

Specific component-technology-development programs and an overall demonstrator-engine program are recommended in order to advance the small transport aircraft propulsion system technologies to a level of acceptable readiness for commercial development by 1988. The overall program approach entails the integration of the several component programs with the demonstrator-engine program. The scope of the recommended program is shown in Table XXIV, and the schedule for the major program elements is given in Figure 31. The program follows the traditional sequence, commencing with the definition and design of the baseline engine to incorporate the advanced-technology components. Concurrently, the advanced-technology components will be designed and tested in full-scale component test rigs. Subsequently, HP-spool components will be installed in the gas generator for further evaluation. Since the bulk of the recommended advanced-technology components are gas-generator components, these items are critical to the overall successful demonstration of the STAT engine. Similarly, following separate component rig tests, the LP-spool components will be combined with the gas generator to make up the complete demonstrator engine, and additional tests will be performed to demonstrate technology readiness for full-scale commercial development. Propulsion-system an ',sis will be performed throughout the program to provide a clear and continuous understanding of the relationship between the propulsion system design tradeoffs and overall airplane performance and costs.

5.2 Preliminary Design

The preliminary-design task will establish the baseline configuration of the complete demonstrator engine. The engine cycle will be defined and component sizes will be established to set the design requirements of each of the STAT advanced-technology components. The demonstrator-engine design will be based on a front-drive, concentric-shaft, free-turbine, turboprop engine configuration, as shown in Figure 32. The nominal takeoff power rating for the engine will be approximately 1350 kW (1810 hp).

Design objectives will be established for each of the advanced-technology components. These objectives, including performance, weight, and manufacturing cost, will be compatible with the overall engine technology required. The demonstrator engine will be designed for ground testing only, and would not necessarily have flight-weight or production-type components in all areas. Components that do not involve the development of new technology will be designed for maximum program economy, while ensuring that the engine will be a representative demonstrator for both steady-state and dynamic operation.

TABLE XXIV. STAT EXPERIMENTAL PROGRAM SCOPE.

- Baseline Engine Definition
- Advanced Technology Component Development
  - PM Titanium Centrifugal Compressor
  - 20:1 P/P Axial/Centrifugal Compressor
  - Single-Stage HP Turbine
  - HP-Turbine Tip Treatment
  - HP-Turbine Active Clearance Control
  - LP-Turbine Active Clearance Control
- Gas-Generator Development
- Demonstrator-Engine Development
- Propulsion-System Analysis

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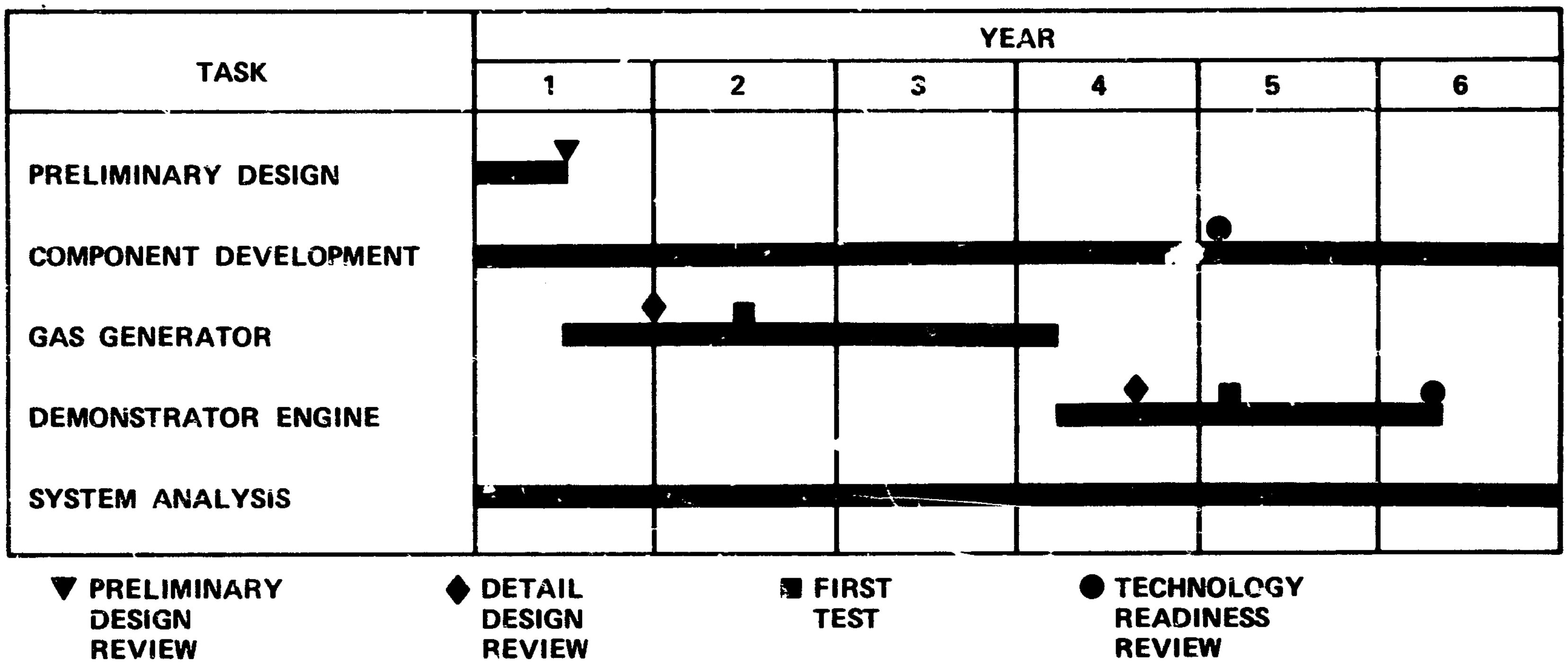

Figure 31. Recommended STAT Experimental Program Schedule.

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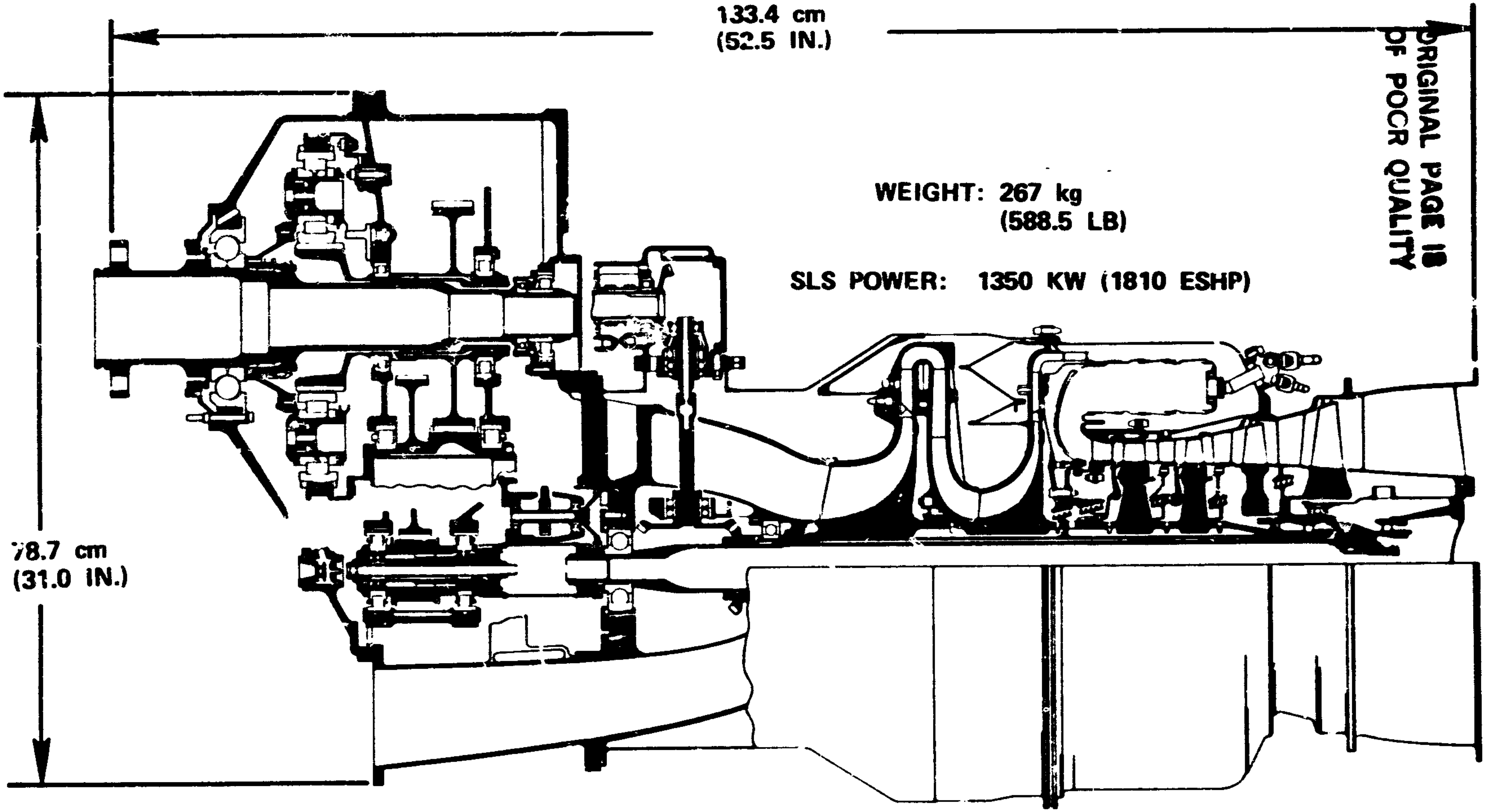

Figure 32. STAT 1990 Advanced Technology Engine.

5.3 Component Technology

As a result of the advanced-technology screening, two gas-generator configurations were identified that showed benefit with respect to the baseline configuration. The first configuration substitutes a single-stage HP turbine for the two-stage HP turbine, and retains the 16:1 two-stage centrifugal compressor. The second configuration substitutes a 20:1 pressure ratio axial/centrifugal for the 16:1 two-stage centrifugal compressor, but retains the two-stage gas-generator turbine. A two-stage turbine is required to drive the 20:1 compressor. Major uncertainties exist in the following areas:

- Foreign-object-ingestion tolerance of the centrifugal versus the axial/centrifugal compressor
- Manufacturing cost of the axial/centrifugal compressor
- Variable-geometry requirements of the axial/centrifugal compressor
- Shaft dynamics.

Another facet is the question of fuel cost. The tradeoff studies reported herein assumed $0.264/l ($1.00/gal) fuel cost. Fuel-cost increases would favor the axial/centrifugal compressor and the two-stage turbine as their benefit is attributable to higher efficiencies. Further study is required to select an approach, and would have to include detail design and manufacturing studies.

In order to define a program for the compressor and turbine, and to enable a benefit analysis to be conducted, programs for highest-risk compressor and turbine were defined; i.e., the axial/centrifugal compressor and the single-stage turbine. These programs are representative of two-stage centrifugal compressor and two-stage turbine programs; although it is expected that program cost would be slightly less and could be accomplished in a shorter period of time.

5.3.1 Compressor

Two compressor-technology programs are recommended:

- 20:1 pressure ratio axial/centrifugal compressor
- PM titanium centrifugal impellers.

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5.3.1.1 20:1 Pressure Ratio Axial/Centrifugal Compressor

The objective of the 20:1 pressure ratio axial/centrifugal compressor program is to provide the technology for improving engine cycle efficiency with the higher cycle pressure ratio, while increasing the overall component efficiency over that of the 16:1 two-stage centrifugal compressor baseline design.

The program schedule for the 20:1 pressure ratio compressor design and evaluation is shown in Figure 33. The program is a 4-year, three-phase effort that provides for design modifications to optimize performance over a broad speed range. The initial design phase for Build 1 will include a preliminary design to define gas flow path, stage work split, stage pressure ratios, and basic mechanical data required for the test rig design. Detail design will include definition of blade and vane contours, aerodynamic loadings and losses, and disk and blade stresses and vibration characteristics. Within this same time period, design modifications of an existing test rig will be completed, including the definition of variable-geometry features and instrumentation.

The initial test sequence will involve testing of the axial and centrifugal compressors separately, followed by an axial/centrifugal compressor test. It is anticipated that all vane rows in the axial compressor will be movable to provide for rapid optimization of the design without time-consuming teardowns and rebuilds. This variable-vane feature will also be utilized to vary off-design performance characteristics of the axial compressor to optimize part-speed efficiency. The centrifugal stage will be tested with the inlet duct, simulating conditions from the axial compressor.

Design modifications required from the first series of tests will be defined and fabricated. The second series of tests will again be the individual axial and centrifugal compressors, followed by an axial/centrifugal compressor test.

Final design modifications will follow this test series. The final test will only incorporate variable geometry in the axial stages required for good part-speed performance. The tests will be a complete mapping of the axial/centrifugal compressor to demonstrate performance goals.

5.3.1.2 Powder Metallurgy Centrifugal Impellers

The use of PM titanium for centrifugal impellers offers the potential of a 25- to 40-percent reduction in component cost without compromising performance or affecting engine weight. The cost reduction is predicated on the successful development of fabrication processes that will permit the production of centrifugal impellers in net shape, thus eliminating the complex and time-consuming machining of the impeller blades. It is anticipated

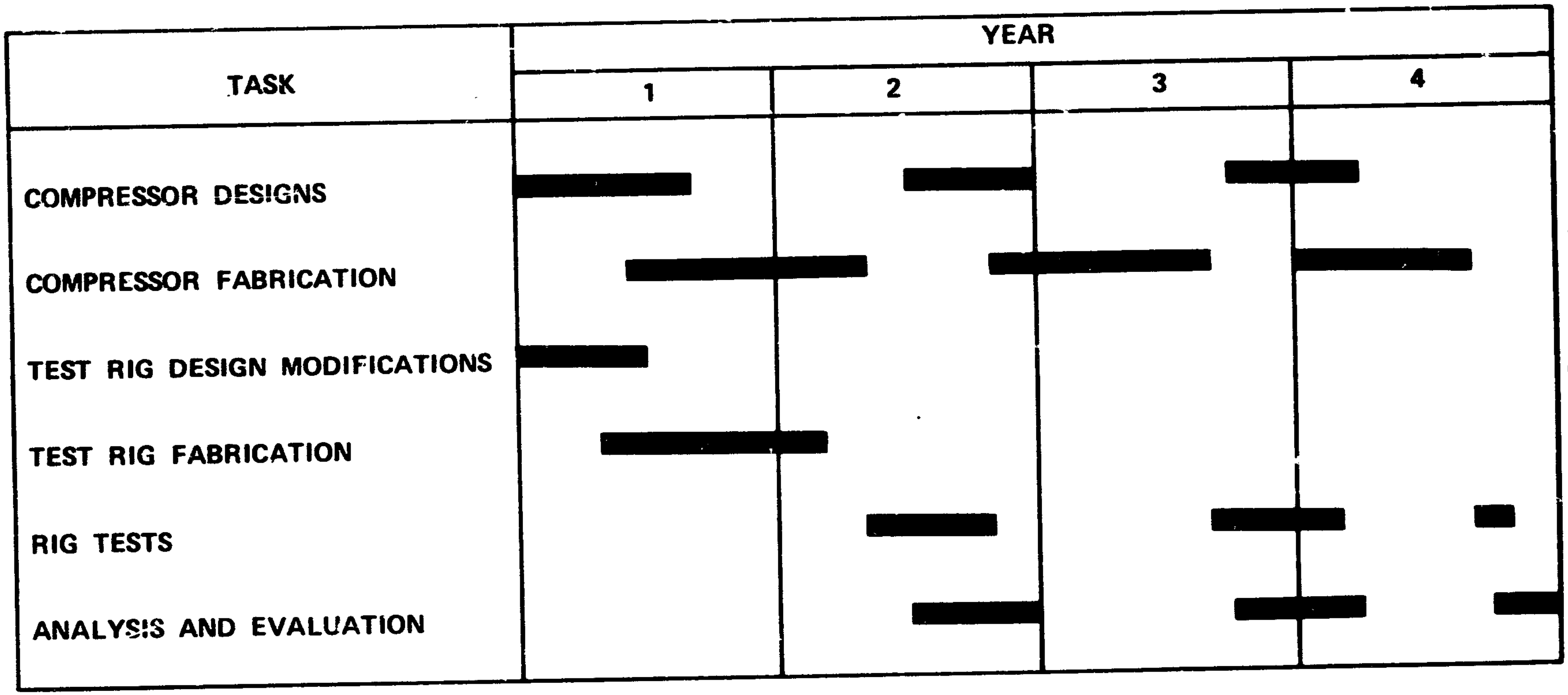

Figure 33. 20:1 Pressure Ratio Axial-Centrifugal Compressor Program Schedule.

that a PM titanium impeller compressor could be used either in the first- and second-stage of a two-stage centrifugal, or as the centrifugal component of the 20:1 pressure ratio axial/centrifugal compressor, discussed in the preceding section. In the latter case, the cost benefit mentioned above would not be as great.

The objectives of the PM titanium compressor program are to verify the potential production cost advantage of the PM process and to determine whether any peformance penalty results from producing net-shape impellers with this process.

The program schedule for the PM titanium compressor program is given in Figure 34. The program is a 32-month, seven-task technical effort that draws on the materials technology currently being developed by Garrett under U.S. Army sponsorship. The program outlined in Figure 34 includes the design of the PM impeller and its manufacturing tooling, and the testing of a conventional 3-D impeller machined from a titanium forging, followed by a comparison of the aerodynamic test data and an evaluation of the manufacturing costs of both impellers.

The results of the performance and cost comparison will be used to determine if any changes to the design or fabrication methods are required prior to initiating gas generator tests.

5.3.2 Turbines

The HP turbine for the 1990 baseline engine is a two-stage, cooled, axial configuration with forged machined hubs, cast inserted directionally solidified (DS) Mar-M 247 blades, passive clearance control, and flow discouragers. The LP turbine is a two-stage axial configuration with a cooled first-stage stator, forged machined hubs, cast inserted uncooled DS Mar-M 247 blades with integral shrouds, and passive clearance control. Of the 16 advanced-technology features examined for these turbines, the following features are recommended for further research and development.

- <u>Single-stage HP turbine.</u> - Results in reduced engine cost and weight, and slightly increased SFC, with a 1-percent reduction in airplane DOC (16:1 pressure ratio only).

- <u>HP-turbine tip treatment.</u> - Results in reduced rotor tip leakage and improved efficiency, no impact on weight or cost, with a reduction in airplane DOC.

- <u>HP-turbine active clearance control.</u> - Results in potential efficiency increase at the expense of an additional cooling-air penalty and cost increase and an overall reduction of airplane DOC.

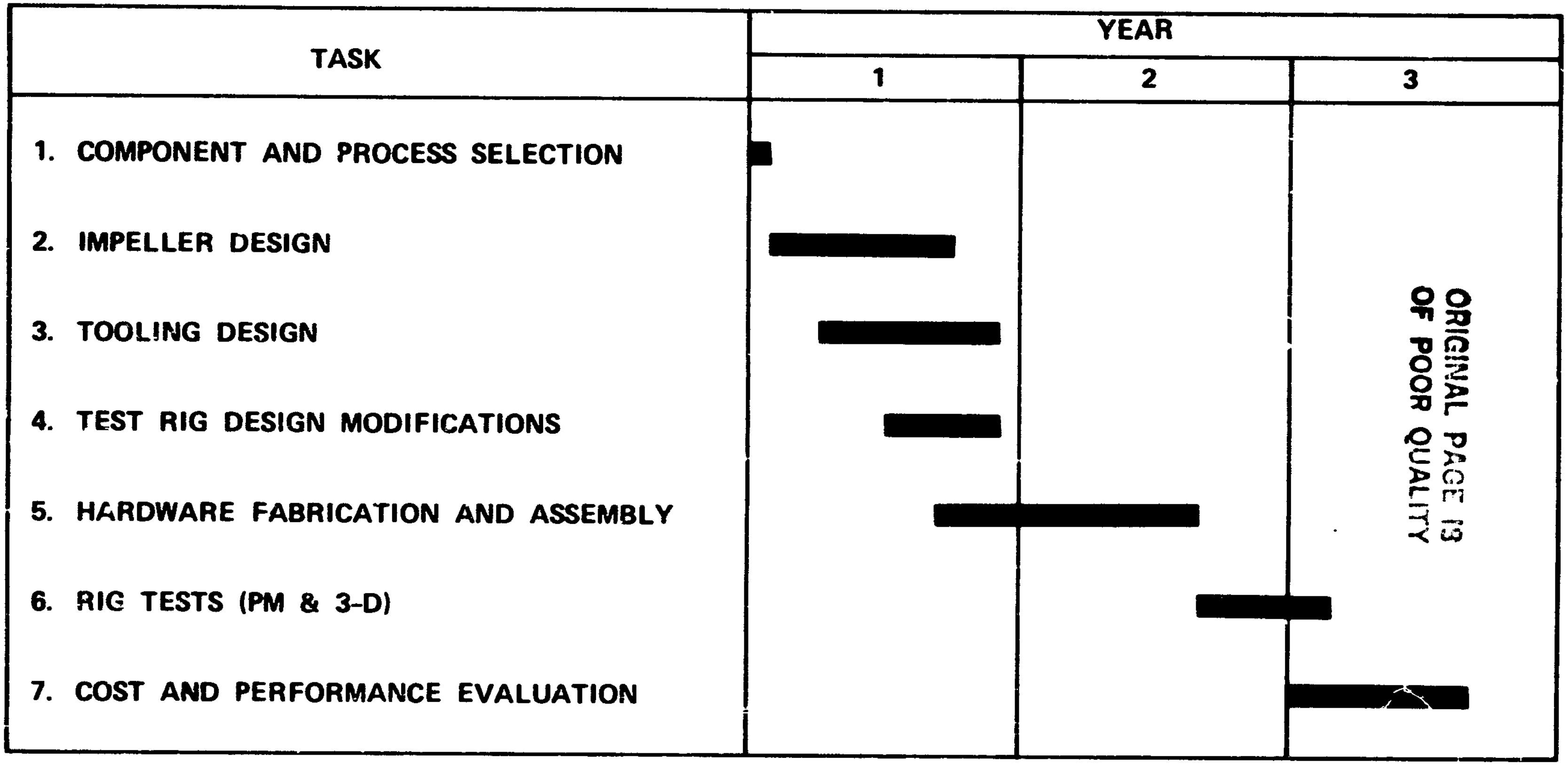

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Figure 34. PM Titanium Centrifugal Compressor Schedule.

- Turbine cooling flow modulation. - Results in lower cooling and flow and lower SFC at cruise power.

- LP-turbine active clearance control. - Results are similar to those for HP turbine, with a somewhat greater reduction anticipated in airplane DOC.

5.3.2.1 Single-Stage HP Turbine with Tip Leakage Control

The incorporation of a single-stage HP turbine offers the advantages of reduced engine weight and cost with only a slight efficiency penalty. In order to offset that penalty, it is considered desirable to include means for controlling tip leakage in order to achieve an efficiency improvement. The program outlined below includes the design and optimization of a 3-D, high-work, single-stage, cooled, HP turbine with tip and shroud treatments to minimize losses due to tip leakage. This program extends the work already accomplished by Garrett and others with high-work, low-aspect-ratio turbine (LART) stages and tip-leakage control methods. This work includes the design and test of a LART stage, and the modeling and testing of tip and shroud treatment methods in cascade rigs and turbine rigs.

The objectives of the single-stage HP turbine program are: (1) to design and test a high-work, cooled, single-stage, HP turbine that is optimized for minimum tip-leakage loss by the use of blade-tip and casing treatment; and (2) to verify the cost and weight advantages of the single-stage over a two-stage design with respect to airplane DOC.

The program schedule for the single-stage HP turbine program is shown on Figure 35. This program is a 3-year effort that involves: (1) the design of a baseline high-work turbine stage; (2) analysis and evaluation of various turbine-tip and shroud treatments leading to the optimization of the turbine blade configuration; (3) testing of the optimized blade stage; (4) analysis of the stage and treatment performance; and (5) comparison of the weight and cost of a production configuration with a two-stage turbine. At the conclusion of this program, gas generator tests would be initiated.

5.3.2.2 Turbine Active Clearance Control and Cooling Air Modulation

Clearance-control schemes can result in improved engine performance by reducing the turbine tip losses. However, these schemes can also increase the complexity and cost of the engine by introducing additional hardware and control functions. In addition, they can impose a cycle penalty since they employ bleed air. The active clearance-control system recommended for STAT is an ON-OFF system that would only be utilized during the cruise mode.

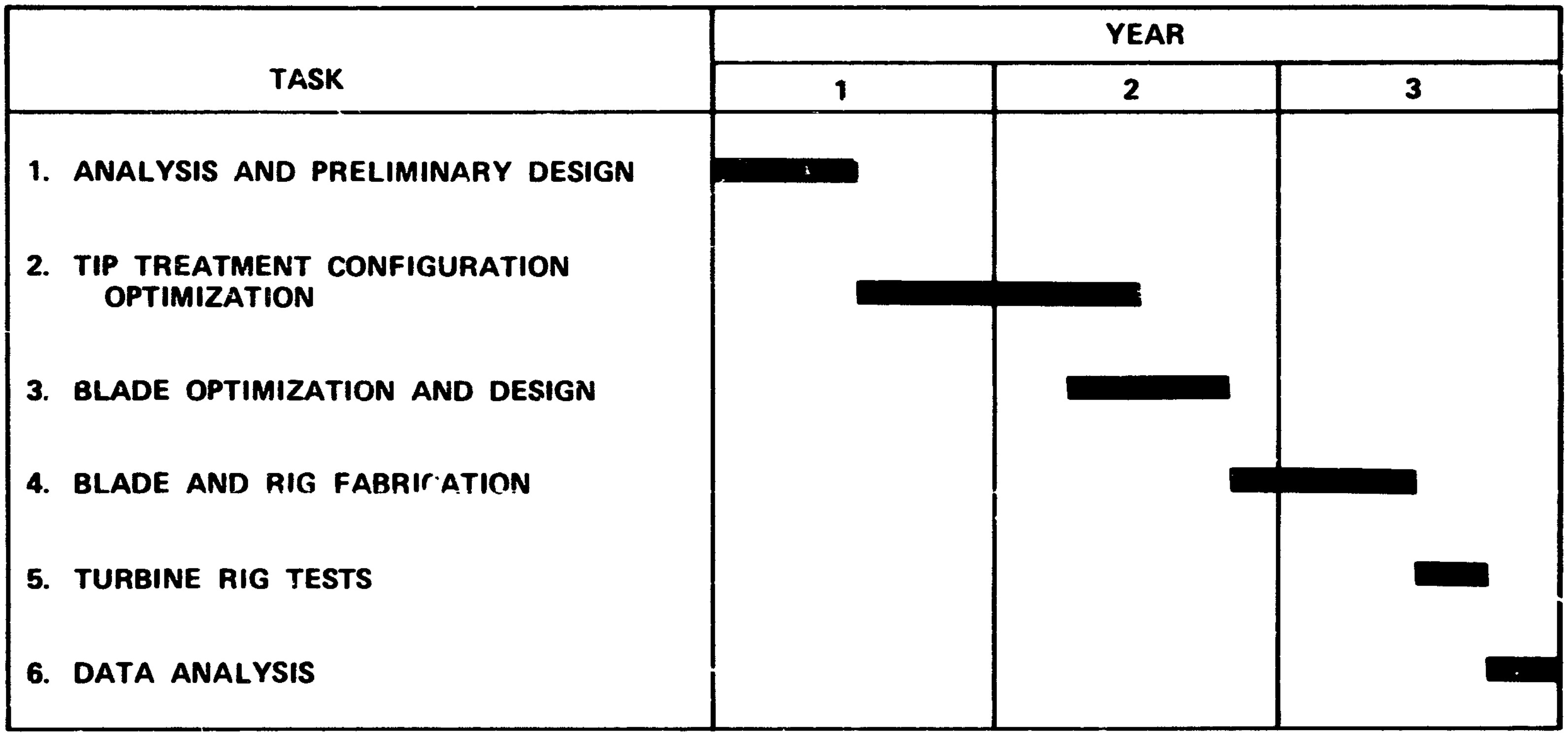

Figure 35. Single-Stage HP Turbine with Tip Leakage Control Schedule.

In this system, interstage bleed air is manifolded to the turbine casing through an on-off solenoid valve. During takeoff and climb the valve is closed, allowing therm expansion of the casing. At start of cruise, the valve is opened, permitting relatively cooler bleed air to impinge on the turbine outer structure, which contracts to reduce the turbine tip clearance.

The objectives of the turbine active clearance control program are to demonstrate the capability for controlling HP- or LP-turbine tip clearance and to define the cycle, weight, and cost penalties associated with incorporating such a system on a production STAT engine.

The program schedule for the turbine active clearance-control program is shown in Figure 36. The program is a 12-month technical effort that includes the definition and design of one of several configuration alternatives, fabrication of hardware, modification of engine components, full-scale engine testing, and evaluation of data. The engine tests would include operation over a range of bleed-airflow rates to determine the relationship of the clearance-control effectiveness to the bleed-air cycle penalty.

Various concepts for modulating the amount of cooling flow to the turbine have been proposed. It is proposed that the design effort to mechanize a selected concept and analysis required to assure fail-safe operation be combined with the turbine tip treatment task.

5.4 Gas Generator

The gas-generator task is structured to permit early testing of the baseline gas generator and, subsequently, to permit incorporation of advanced-technology components as the component tests are completed. In this manner, problem areas will be revealed early enough to implement corrective action prior to the full demonstrator engine tests. The schedule for the gas generator program is given in Figure 37.

After completion of the engine preliminary design, the detail design of the baseline engine will be initiated. This will include the baseline components for the HP spool, and will also include provision for subsequent modification to incorporate the 20:1 presssure ratio axial/centrifugal compressor and the single-stage HP turbine. This design will also provide the workhorse test engine for the PM titanium impeller and the HP-turbine active clearance control, tip treatment and cooling flow modulation.

Hardware for two complete engines plus spares will be procured so that refurbishment can be accomplished as required during the test program.

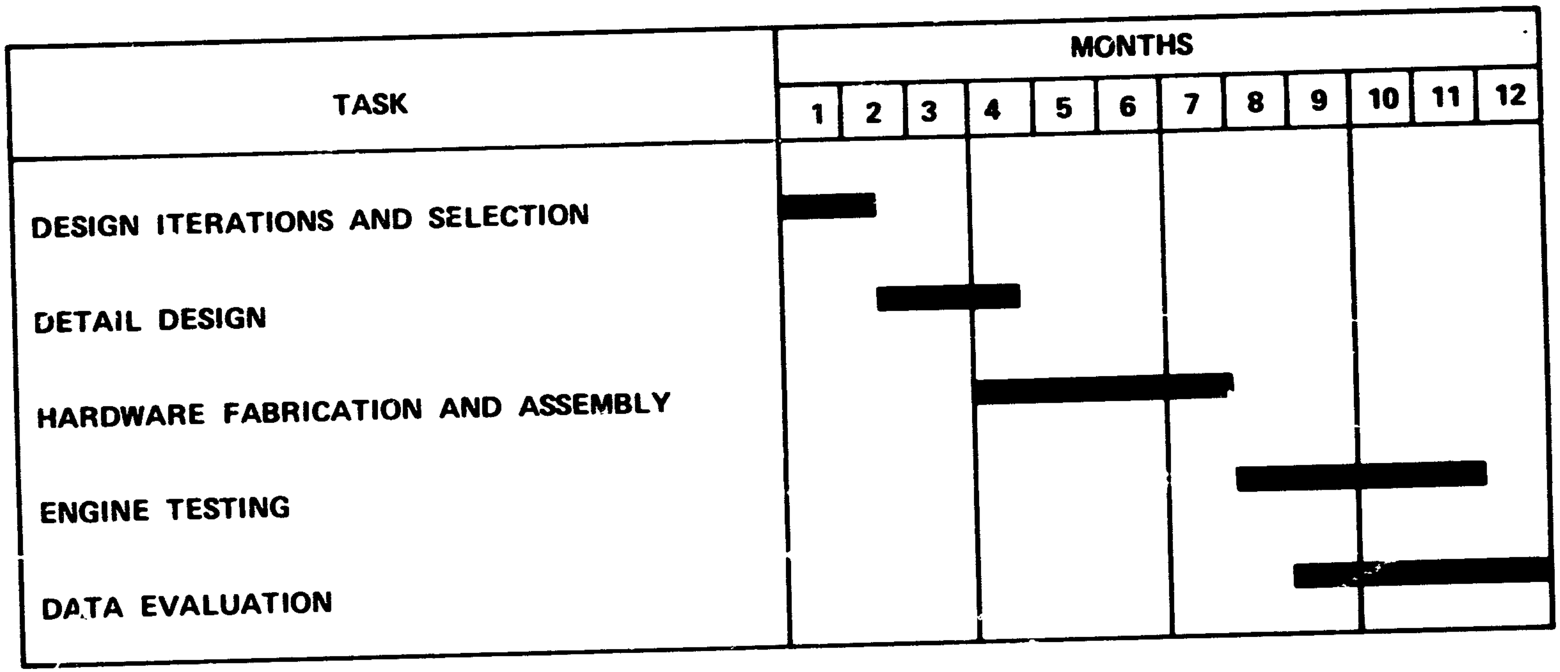

Figure 36. Turbine Active Clearance Control Program Schedule.

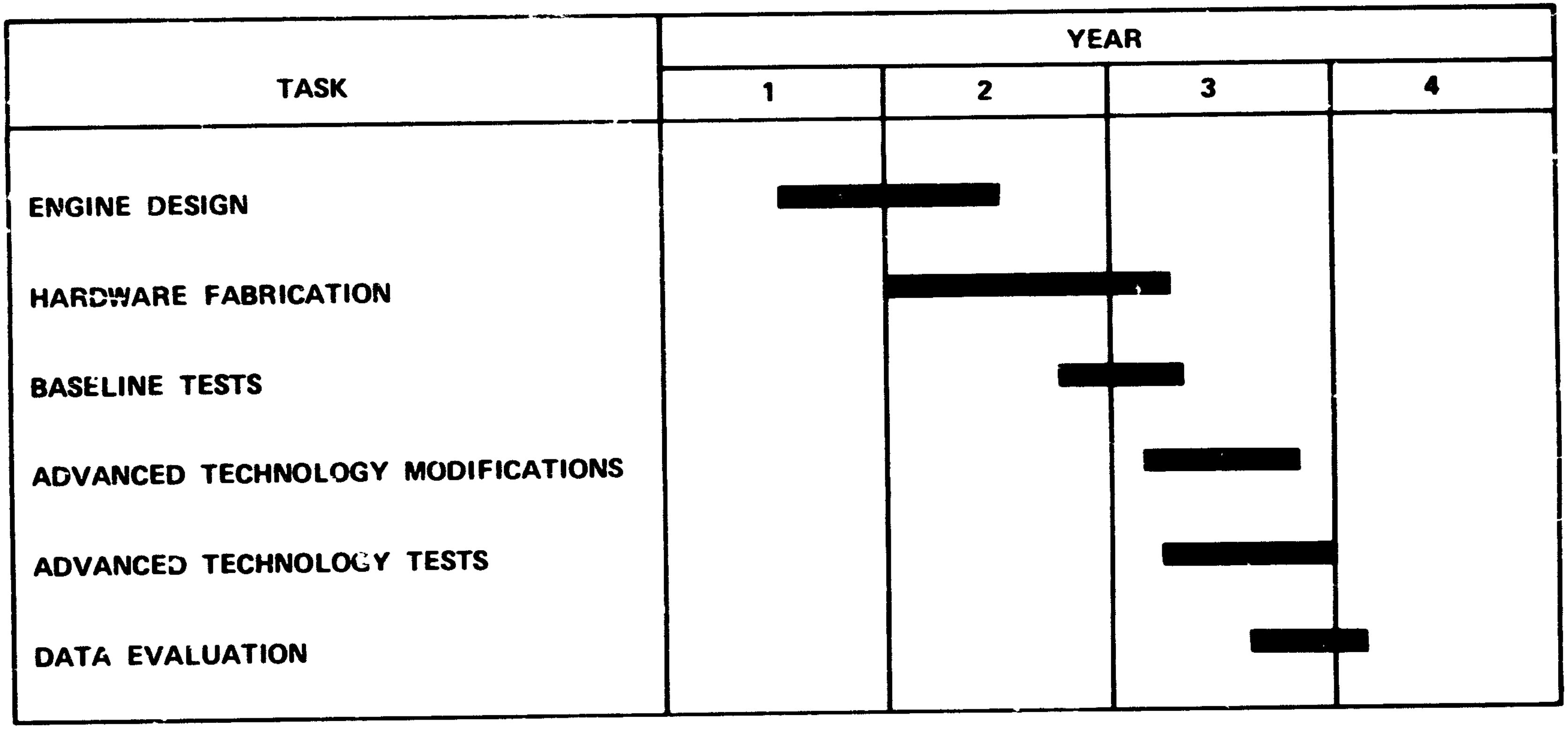

Figure 37. Gas Generator Program Schedule.

The initial tests of the baseline gas generator will include performance checks and a 40-hour endurance test. Previous experience has shown that any significant performance deterioration due to seal wear, differential thermal growth, or similar causes will be revealed after the initial 40 hours of operation. These tests will be completed prior to completion of the advanced-technology component tests in sufficient time to prepare the gas generator for the advanced component evaluations in the engine environment.

Since the gas generator will be used for detailed component performance comparisons, the initial baseline assembly will include extensive instrumentation to monitor engine health and to obtain engine and component data. The baseline tests will include evaluation of the following:

- o Green run (mechanical checkout)
- o Overall engine performance (40-hours endurance)
- o Compressor performance
- o Combustor performance
- o HP-turbine performance.

Following the baseline tests, the gas generator will be disassembled for inspection. Any necessary refurbishing will be accomplished, and the engine will be reassembled to begin the advanced-technology test series. In this series, each of the advanced-technology components will be tested individually so that separate comparisons against the baseline can be made. The advanced-technology component test series will include evaluations of the following:

- o 20:1 pressure ratio axial/centrifugal compressor
- o PM titanium compressor
- o Single-stage HP turbine with tip-leakage control
- o HP-turbine active clearance control.

It is anticipated that a total of approximately 165 hours of test time will be accumulated during the gas-generator tests, of which approximately 50 hours will be accumulated during the baseline test series. At the conclusion of the gas-generator tests, the engine will be disassembled, inspected, and prepared for assembly for the demonstrator engine tests.

5.5 Demonstrator Engine

In conjunction with the gas-generator task, the demonstrator-engine task will provide the means to advance the STAT propulsion systems technologies to the level of acceptable readiness for commercial development by 1988. The demonstrator-engine task couples the gas generator with the LP spool and gearbox to provide a complete experimental test vehicle for STAT technologies. The demonstrator-engine program schedule is given in Figure 38.

Detail design of the LP-spool components commences at the conclusion of the gas-generator program. Advantage can thus be taken of any low-spool modifications to the earlier preliminary design that are indicated from the gas-generator tests. The LP-spool detail design includes the LP turbine, LP shaft and bearings, supporting structure and exhaust duct, and the output gear system and gearbox.

Sufficient demonstrator-engine hardware will be procured to build two complete engines. In additon, spares of critical components (such as the combustor, turbine vanes and blades, and bearings) will be obtained in order to refurbish the engine as required during the test program.

The initial assembly of the demonstrator engine will include the baseline gas-generator components, the LP spool (without clearance control), and the output gearbox. This assembly will include only the instrumentation necessary to monitor the health of the engine. A brief series of baseline tests will be performed during which the operating characteristics of the LP spool will be closely monitored and the overall performance of the engine will be measured. The baseline tests include a green run, during which the mechanical operation and vibration characteristics of the engine are checked; a 40-hour endurance test, during which performance at various powers from idle to full-load is measured; and throttle-response checks.

Following the baseline tests, the engine will be disassembled for inspection and reassembled with instrumentation required to measure the LP-turbine performance with active clearance control. The active clearance-control test is the only advanced-technology component test specifically applicable to the LP spool, and will therefore be conducted as a separate test prior to conducting additional advanced-technology tests.

The demonstrator-engine program has been structured to accommodate an additional series of advanced-technology tests. These tests will be specified following the gas-generator tests, and can include any of the features tested with the gas generator or other features that may be considered beneficial. Among these optional features are:

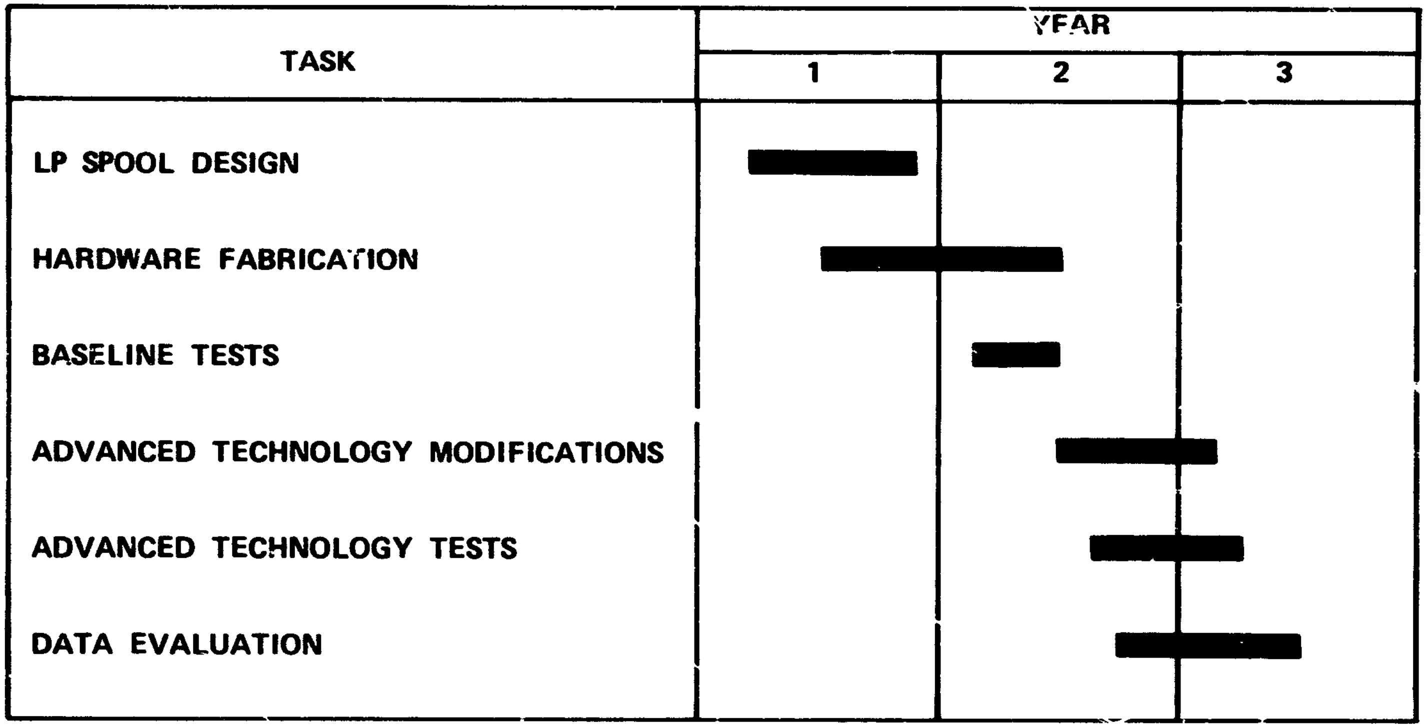

Figure 38. Demonstrator Engine Program Schedule.

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- Uncooled, single-crystal, HP-turbine blades
- Single-stage LP turbine
- Higher temperature turbine disks
- High effectiveness turbine-cooling schemes.

Each of the features selected for the optional advanced-technology tests will be evaluated individually against the baseline configuration for their effects on overall engine performance and for their ultimate effect on airplane DOC.

A total of 150 hours is expected to be accumulated during the demonstrator-engine tests, of which approximately 50 hours will be accumulated during the baseline tests.

5.6 System Analysis

As indicated in Figure 31, system analysis will be conducted throughout the STAT experimental program. These analyses will include integration of updated component data into the engine performance model, and incorporation of engine and component detail design and test data into the engine cost, weight, life, and maintenance characteristics. This updated engine data will also be used in combination with current airplane-sensitivity data to re-evaluate airplane DOC. In these analyses, particular emphasis will be given to the accurate assessment of acquisition and maintenance costs, since, in addition to fuel costs, these are prime considerations to the aircraft owner/operator.

5.7 Benefit Analysis

A benefit/cost analysis was performed for each of the component technology programs described in Section 5.3, above. These analyses were performed for the 30-passenger, 1990 baseline airplane operating over a 100-nmi mission. To devise a baseline benefit/cost ratio, a fleet size of 250 airplanes was assumed, operating for a period of 5 years with a utilization rate of 2500 hours per year and fuel cost of $0.264/1 ($1.00/gal). The benefit/cost analyses were based on the ratio of the cost savings anticipated for the incorporation of each individual technology into the 1990 baseline engine to the anticipated development cost to bring that technology to a level of acceptable readiness for commercial development by 1988. The development cost used as the denominator of the benefit/cost ratio is the development cost over and above the cost to develop the equivalent component in the 1990 baseline design. For example, in the case of the 20:1 pressure ratio axial/centrifugal compressor, the cost used is the difference between the development cost of the 16:1 pressure ratio two-stage centrifugal and the 20:1 pressure ratio axial/centrifugal compressors. The cost/benefit ratio was then modified to reflect its probability of success.

The method for assessing the probability of success is based on a risk analysis method used in other programs at Garrett for NASA and is summarized in Tables XXV and XXVI.

The results of the benefit/cost analyses are shown in Table XXVII. Included in this table are the performance, weight, and cost factors used to determine the effect of each technology on airplane DOC. A cost/benefit ratio of above 3:1 is considered attractive when consideration is given to the payoff period assumed (5 years), the fleet size (250 A/C), and the price of fuel used [$0.264/1 ($1.00/gal)]. All of the advanced-technology components exceeded a cost/benefit ratio of 3:1 with the exception of the PM titanium compressor fabrication approach.

The benefit/cost analysis was extended to longer operational periods, a large fleet size, and higher fuel costs.

Specifically, the effects of a parametric variation of these conditions; i.e., increase the fleet size to 1000 airplanes; increase the time period to 20 years; and increase fuel cost to $0.528/1 ($2.00/gal) were determined. The results, in the form of a relative benefit/cost ratio, are shown in Figure 39 for the 30-passenger, 1990 baseline airplane operating over a 100-nmi mission. This figure shows, as would be expected, that as each of the parameters in increased, the relative benefit for a given technology also increases; and as the normal life of an airframe is approached (say 20 years), the relative benefit/cost ratio approaches a 10-fold increase.

TABLE XXV. TECHNOLOGY DEVELOPMENT RISK ASSESSMENT FACTORS.

	Degrees of Risk		
	A	B	C
Primary Factors			
1. Nature of Technology	Traditional	Advanced	Revolutionary
2. Design Approach/Application of Technology	Traditional	Advanced	Revolutionary
3. Current Status of Technology	Production Feasibility	Component Feasibility	Laboratory Feasibility
Secondary Factors			
4. Number of Alternative Approaches for Application/ Opportunities of Increment Success for Technology	3 or More	2	1
5. Required Technology Incorporation Date of Technology (years)	7	5	3
6. Critical Nature of Component to which Technology is Applied	Static/Low Stress	Static/High Stress	Rotating

TABLE XXVI. PROCEDURE FOR QUANTITATIVE RISK ASSESSMENT.

Baseline Risk Assessment (Primary Factors 1 through 3)

- Degrees of risk for first 2 factors is A, minimum criterion for 90% Probability of Success:
  - +Either A or B for 3rd factor → 90% Probability of Success.
  - or +C for 3rd factor → 75% Probability of Success.
- Degree of risk for first 2 factors is B, minimum criterion for 75% Probability of Success:
  - +Either A or B for 3rd factor → 75% Probability of Success.
  - or +C for 3rd factor → 50% Probability of Success (75% if either other factor is A).
- Degree of risk for first 2 factors is C, minimum criterion for 50% Probability of Success:
  - +Either A or B for 3rd factor → 50% Probability of Success.
  - or +C for 3rd factor → 25% Probability of Success (50% if either other factor is A or B).

Refinement for Baseline Assessment Result of 25% Probability of Success or Two Primary Factors C (Secondary Factors 4 through 6).

- Probability of Success values <25% are subdivided into two scale levels, 10% and 0%.

o Degree of risk for 1 secondary factor is C	No change in Probability of Success.
o Degree of risk for 2 secondary factors is either B or C	Probability of Success reduced one level.
o Degree of risk for all secondary factors is either B or C	Probability of Success reduced two levels.

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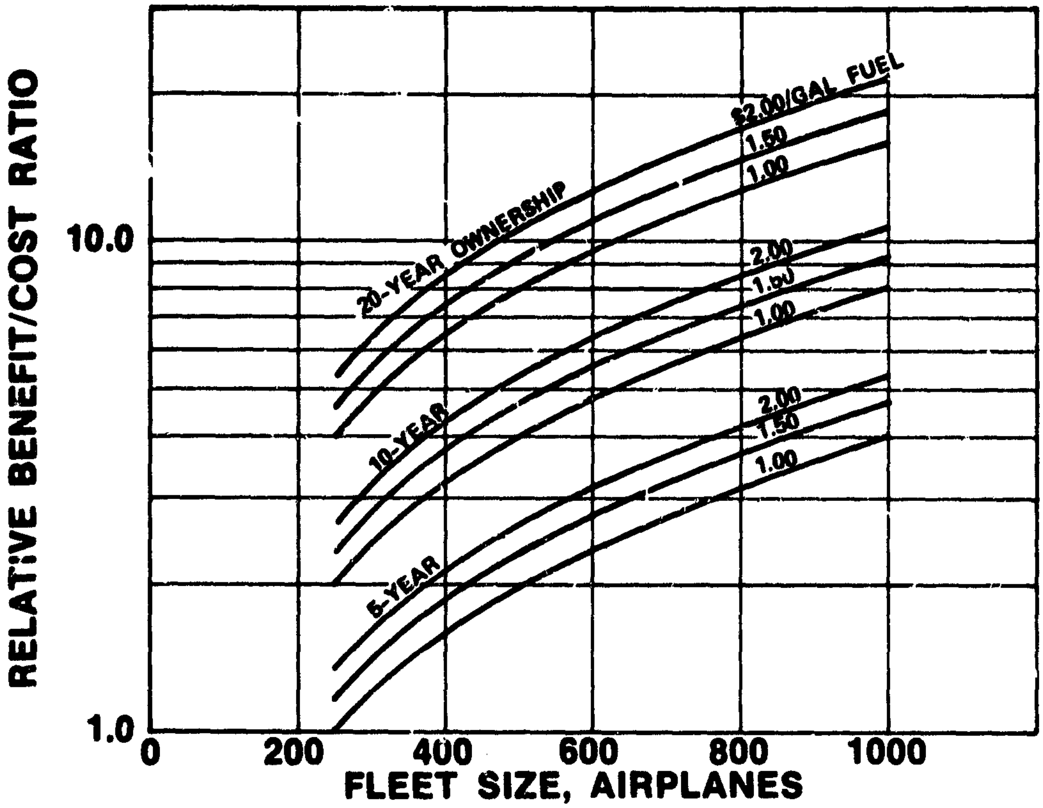

Figure 39. Relative Benefit/Cost Ratio, 30-PAX 1990 Baseline Airplane, 100-nmi Mission.

TABLE XXVII. STAT ADVANCED TECHNOLOGY COMPONENTS BENEFITS.

[1990 BASELINE AIRPLANES, 100-NMI MISSION, \$0.264/l (\$1.00/GAL) FUEL AND 1990 ADVANCED TECHNOLOGY ENGINE, CONFIGURATION A]

	$\Delta\eta$ POINTS	ENGINE IMPROVEMENTS			% DOC		BENEFIT/COST RATIO	
		ΔSFC %	ΔWEIGHT %	ΔCOST %	30-PAX	50-PAX	30-PAX	50-PAX
Powder Metallurgy Titanium Centrifugal Compressor	NC	NC	NC	-0.7	-0.22	-0.12	1.2	1.0
Single-Stage High Pressure Turbine	-2.0	-1.8	-12.4	-8.6	-1.21	-0.77	6.4	5.7
High-Pressure Turbine Tip Treatment	+1.0	-0.8	NC	NC	-0.40	-0.42	4.8	7.0
High-Pressure Turbine Cooling Flow Modulation	NC	-0.8	NC	NC	-0.40	-0.42	11.2	16.5
High-Pressure Turbine Active Clearance Control	+1.0	-0.9	NC	NC	-0.43	-0.35	18.1	20.6
Low-Pressure Turbine Active Clearance Control	+1.0	-1.2	NC	NC	-0.58	-0.58	24.4	34.2
Laser Hardened Gears	NC	NC	NC	-1.8	-0.13	-0.11	*	*
Photo Etched Combustor	NC	NC	-0.2	-0.6	-0.15	-0.12	0.8	0.9

NC - No Change

*Will be available through normal industry development.

6.0 CONCLUSIONS AND RECOMMENDATIONS

This report summarizes the results of the Small Transport Aircraft Technology (STAT) Program. Turboprop engines in the 1200 to 1800 kW (1600 to 2400 hp) range were defined and evaluated in 30- and 50-passenger airplanes defined by NASA-Ames. The performance, weight, and cost effects of these engines were evaluated on the DOC of the airplanes. In addition, the effects on the engine performance, weight, and cost, and on the airplane DOC of a series of advanced engine technologies were evaluated, with the result that development programs for several of these technologies are recommended for incorporation in an overall engine development program for small transport aircraft.

Overall conclusions drawn from the STAT study program are:

- Advanced turboprop engines offer a 21- to 23-percent decrease in airplane DOC relative to 1980 production engines.

- Incorporation of engine design features tailored to commuter requirements contribute to lower maintenance cost.

- Reduced mission fuel consumption and lower maintenance costs are primary contributors to the reduced DOC.

- High-cycle technology (high pressure ratio and high turbine temperature), even though more costly than low-cycle technology, is superior with respect to overall operating cost.

- The large potential benefit of a 1990 advanced turboprop engine relative to 1985 derivative engines ($\Delta \cong 10\%$ DOC) provides strong encouragement for NASA to sponsor advanced research and development efforts oriented toward commuter propulsion systems.

APPENDIX I

COMPUTER PRINTOUTS OF AIRPLANE DEFINITIONS
(FURNISHED BY NASA-AMES RESEARCH CENTER)

- STAT 30-PAX Airplane Definition
- STAT 50-PAX Airplane Definition

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DATE  10/21/80                                    STAT 30PAX/TPE331-11
INPUTS - CONFG  WG = 29554.8   W/S = 60.0   PAX=   30.0   EMCRU =   .455   HMCRU = 10001.0   ELN =13.393   DBARN =   3.7
         WING   TCT =   .122   TCM = .182   AR =  10.00   SLM   =   .300   DLMCA =   5.000   EVE = 2.000   ALPHLUB =1.50
         FLAPS  DFLPTO= 25.0   DFLPLD=50.0  FELLEC= 0.0   DCLPTE= 1.000   DCDVTE=  .13000   CRF = 2.896   CRF   = 1.623
         ENGINE NO.ENGS=  2.0  ACCECR= 4.0  XNMAX=30771.  GM    =  .0354  XTORG = 99999.0   ASFC  =1.000  ANCUMP= .0570
         PROP   PCNCCP= .960   DPROP=12.45  BL  =   5.0   TSPDMX= 722.0   AF    = 116.000   CLI =   .508  PCLERE  .3863
         WGHTS  SKPEI= .2500   SKSLS= .230  UNNAC=2.100   XLOCE =   .360  DBGUAL  =  85.0   SKPES= .338   SKY   =   .170
                SKZ  =  .0000  SKWW =133.4  SKR  =125.0   SKSAS =    0.0  DELP  =    5.00   SKPFAC= .80   STATIC=   .038
         MISSON DELTVR=3.500   DV1 = 10.00  DVR =   5.00  ST ABN=    4.0  XLDNG = 99999.0   FCSF= 1.000   KNAC =     1.0

CONFIG - PAX = 30.0   RANGE = 600.6   WGROSS = 29554.8   WING LOADING =  60.0

ENGINE - HPMSLS = 1039.3   T/W =  .028   HPMCR =  1039.   HPMTO =    0.   HPMCL =    0.  HPMSC =    0.  WEIGHT = 700.3

PROP   - DPROP = 12.05   TIPSPD = 722.0   NO. BL = 5.0   AF =110.0   CLI =  .500   WEIGHT = 434.3   IDATE =  1978.
         PROP RPM = 1090.1   CRUISE PERCENT POWER =  .799   CRUISE PERCENT RPM =  .985  NAC LENGTH,DIAM. =  13.39   3.67

WEIGHTS- ENG WTS = 1520.5   PROP WTS =  868.0   NAC+PLYN WTS =  648.0   EMPTY WT = 18761.7   OWE = 19671.6
         PAYLOAD =  6000.0   FUEL WT =  3883.3 STRUC WT = 11100. FIX EQ. WT =  4339.   ACT TREAT WT = 1889.9

AERODYN- WING AREA =  492.6   ASPCT RATIO = 10.0   TAPER RATIO =  .300   SWEEP =  5.0   HTAIL AREA = 107.2   VTAIL AREA = 100.2
         F.P. AREA =  12.3   CDO =  .02603   CDAMP =  .00301   GSPLD FTR )  .7572   CRUISE CD =  .0250 +   .0397 CLE

                           *** MISSION PERFORMANCE ***

TAKEOFF- HPORY =    0.0   DTEMP = 30.0   MAX CL = 2.24   ACCEL-STOP = 4191.8   CEI DIST TO 35 = 3744.3  V1 =  98.0  VR = 103.0
         AEO DIST TO 35 = 3215.0  FAR25 TO DIST = 3697.3  V35/VSTALL =1.2208  MIN CNTRL SPEED = 93.2  BFL = 3968.0  L/DOV2= 11.11
         SECOND SEGMENT CEI R/C (R/C REQ) = 555.1 (267.5) FPM   CEI APPROACH GOAROUND R/C (R/C REQ) = 380.0 (278.7) FPM
         AT END TO - TIME =  .102   FUEL = 132.7   WEIGHT = 29422.1   ALT =  600.0

LANDING- MAX CL = 2.35   FAR DIST = 4178.3   LND WT = 29554.8   APP. SPEED = 112.7   STALL SPD =  86.7   LNDG W/S =  60.0

CLIMB  - REQ S.C. =     0.0  REQ R/C =    0.0  SER. CLNG = 17522.6  BEST R/C SPEED = 177.5  EAS CLIMB SPEED = 228.6  L/DOSMO 11.02
         AT END CLMB - TIME =  .573   FUEL = 769.9   WEIGHT = 28785.0   RANGE = 110.0   ALTITUDE =20000.0
                       TIME =  .312   FUEL = 408.8   WEIGHT = 29146.0   RANGE =  37.0   ALTITUDE =12000.0

DESIGN
CRUISE - ALTITUDE = 20000.0   FUEL AVAILABLE = 3883.3
                       TIME      RANGE     SPEED      MACH      L/D     SPC RNG     MTCS     ETAP
         NORMAL POWER  2.303     600.6     207.0      .3523    12.247    .24505     284.3    .0049DEG.PT
         BEST SPEC.RNG 2.932     674.5     171.1      .3010    14.057    .25493     230.3    .0000
         INPUT SPEED   3.233     682.6     157.2      .3500    14.292    .25026     215.2    .0037
OFF DES- ALTITUDE =  12000.0   FUEL AVAILABLE = 3883.3
         NORMAL POWER  1.787     458.7     239.5      .4536    13.469    .20065     287.7    .0798
         BEST SPEC.RNG 2.758     583.0     180.2      .3920    13.557    .21930     283.0    .0121
         INPUT SPEED   2.266     532.5     211.2      .4000    12.164    .21620     253.7    .0691

DESCENT- FROM 20000. FT - INCR. TIME =  .272 HR  INCR. FUEL = 265.8  INCR. RANGE =  75.8
         FROM 12000. FT - INCR. TIME =  .093 HR  INCR. FUEL =  74.4  INCR. RANGE =  24.4

COSTS  - A/C COST = 3409883.   AIRFRAME COST = 2045125.   COST/ENGINE = 229917.   COST/PROP = 17703.   MANU TIME =   0.7
         RANGE   B.FUEL   TIME   DOC-$/T   DOC-$/BM   DOC-$/NM   DOC-C/SM  B.E.PAX   B.E. LF   ALTITUDE   CR MACH
          180.    1095.   .851    453.82    533.40     2.910     7.0072     9.50    31.920    20000.0     .4485   DESIGN
          601.    2729.  2.303   1220.89    532.69     2.043     5.9212    10.34    34.482                         MISSION

          180.     677.   .540    283.35    525.07     2.834     8.2131     8.79    29.287    12000.0     .4530   OFF DESIGN
          459.    2490.  1.787   1001.03    560.23     2.182     6.3249    10.83    35.847                         MISSION

NOISE  - F.A.R. 36  TAKEOFF =   0.0   SIDELINE =   0.0   APPROACH =    0.0    CONTOUR AREA FOR   0.0 EPNL =  0.000 SQ.MI.
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DATE  10/21/80                                         STAT 50PAX/TPE331-11
INPUTS - CONFG   WG = 43110.5     W/S =  60.0   PAX=    50.0   ENCRU =    .457   WNCRU = 10001.0   ELN =15.275    DBAHN =   8.3
         WING    TCT =   .124     TCH =  .107   AR =   10.22   SLM =      .500   DLMCA =   5.000   EVEN= 2.000    ALPHLOE =-1.50
         FLAPS   DFLPTO= 25.0     DFLPLD=50.0   DELLED= 0.0    DCLMTE= 1.000     DCMOYE=  .13000  CHM = 2.936    CHF  = 1.510
         ENGINE  NO.ENG=  2.0     NOUECR= 4.0   XNMAX=20547.   GR    =  .0321    XTCRG = 99999.0  XSFC =1.000    ANCOMP= .0570
         PROP    PCNCCR= .960     DPROP=15.89   BL  =    5.0   TSPOMA= 708.0     AF    = 115.000  CLI =  .500    PCLEW=  .3000
         WGHTS   SKPLI=  .2500    SMSLS=  .236  UNNAC=2.100    XLODE =   .360    DAGUAL =   85.0  SKPES=  .330   SKY  =   .170
                 SKZ  =  .4000    SKW = 133.4   SKB  =125.0    SKBAS =    0.0    DELP  =    5.00  WKPFAC=  .80   STATIC=  .030
         MISSON  DELTVR=3.500     DV1 = 10.00   DVR =   5.00   ST ABN=    4.0    XLDRG = 99999.0  FCSF= 1.000    XNAC =    1.0

CONFIG - PAX = 50.0   RANGE = 605.5   WGROSS = 43110.5   WING LOADING =  60.0

ENGINE - HPMSLS = 2471.3   T/W =  .426   HPMCR =  2471.   HPMTO =    0.   HPMCL =    0.  HPMBC =    0.  WEIGHT =1039.0

PROP   - DPROP = 15.89   TIPSPD = 708.0   NO. BL = 5.0   AF =115.0   CLI =  .500   WEIGHT = 669.3   IUATE =  1970.
         PROP RPM =  851.0   CRUISE PERCENT POWER =  .799   CRUISE PERCENT RPM =  .986  NAC LENGTH,DIAM. =  15.20   4.25

WEIGHTS- ENG WTS = 2079.6   PROP WTS = 1338.5   NAC+PLYN WTS =  856.6   EMPTY WT = 26767.3   OWE = 27885.0
         PAYLOAD = 10000.0   FUEL WT =  5230.0 STRUC WT = 15447. FIX EQ. WT =  6623.   ACT TREAT WT = 1063.0

AERODYN- WING AREA =  718.6   ASPCT RATIO = 10.2   TAPER RATIO =  .300   SWEEP =  5.0   HTAIL AREA = 291.2   VTAIL AREA = 247.1
         F.P. AREA =  16.3   CD0 =  .02268   CDAMP =  .00362   OSWLD FTR =  .7782   CRUISE CD =  .0227 +   .0400 CL2

                        *** MISSION PERFORMANCE ***

TAKEOFF- HPORT =     0.0   DTEMP = 30.0   MAX CL = 2.28   ACCEL-[illegible] 4226.0   CEI DIST TO 35 = 3827.0   V1 =  97.0   VR = 102.0
         AED DIST TO 35 = 3251.2   FAR25 TO DIST = 3738.0   V35/VSTALL = 1.2198   MIN CNTRL SPEED = 92.5   BFL = 4027.0  L/DOVER 11.90
         SECOND SEGMENT CEI R/C (R/C REQ) = 490.8 (265.3) FPM    CEI APPROACH GOAROUND R/C (R/C REQ) = 380.0 (200.0) FPM
         AT END TO - TIME =  .103   FUEL = 179.6   WEIGHT = 42930.0   ALT =  600.0

LANDING- MAX CL = 2.30   FAR DIST = 4183.0   LND WT = 43110.5   APP. SPEED = 111.0   STALL SPD =  86.0   LNDG W/S =  60.0

CLIMB  - REQ S.C. =      0.0  REQ R/C =    0.0  SER. CLNG = 16895.2  BEST R/C SPEED = 170.0  EAS CLIMB SPEED = 223.0  L/DOBAR 12.03
         AT END CLMB - TIME =  .634   FUEL =1100.0   WEIGHT = 41935.5   RANGE = 120.0   ALTITUDE =20500.0
                       TIME =  .321   FUEL = 571.3   WEIGHT = 42545.1   RANGE =  38.7   ALTITUDE =12000.0

DESIGN
CRUISE - ALTITUDE = 20500.0   FUEL AVAILABLE = 5230.0
                         TIME      RANGE      SPEED      MACH       L/D     SPC RNG      VTAS       ETAP
         NORMAL POWER   2.332      605.5      204.5     .4008     13.377    .18171      282.4     .0047DEG,PT
         BEST SPEC.RNG  2.863      657.9      172.8     .3885     14.861    .18700      238.0     .0771
         INPUT SPEED    3.145      605.7      155.5     .3500     15.023    .18335      214.8     .0031
OFF DES- ALTITUDE =  12000.0   FUEL AVAILABLE = 5230.0
         NORMAL POWER   1.791      458.6      238.8     .4522     11.427    .14594      280.8     .0235
         BEST SPEC.RNG  2.047      568.2      188.8     .3970     14.302    .16993      228.0     .0000
         INPUT SPEED    2.231      523.1      211.2     .4000     13.120    .15741      253.7     .0584

DESCENT- FROM 20500. FT - INCR. TIME =  .291 HR  INCR. FUEL = 383.2  INCR. RANGE =  81.7
         FROM 12000. FT - INCR. TIME =  .093 HR  INCR. FUEL =  99.8  INCR. RANGE =  25.0

COSTS  - A/C COST = 6807158.   AIRFRAME COST = 4138377.   COST/ENGINE = 300917.   COST/PROP = 27373.   MAN. TIME =   0.0
         RANGE   B.FUEL    TIME    DOC-$/T    DOC-$/BM    DOC-$/NM    DOC-C/SM  B.E.PAX   B.E. LF    ALTITUDE   CR MACH
          200.    1509.    .928     683.58     736.60      3.209      5.0890    13.30     26.709     20500.0     .4008    DESIGN
          600.    3700.   2.332    1703.37     730.53      2.813      4.8924    14.26     28.520                          MISSION

          100.     917.    .541     392.15     724.91      3.022      6.0201    12.10     24.320     12000.0     .4524    OFF DESIGN
          450.    3306.   1.791    1377.42     760.91      3.084      5.2230    14.04     29.273                          MISSION

NOISE  - F.A.R. 36 TAKEOFF =   0.0   SIDELINE =   0.0   APPROACH =   0.0    CONTOUR AREA FOR   0.0 EPNL =  0.000 SQ.MI.
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APPENDIX II

SYMBOLS AND ABBREVIATIONS

A/C	Aircraft
APU	Auxiliary Power Unit
BH	Block Hour
C/L	Centerline
CRS	Commercial Rapid Solidifying
DB	Diffusion Bonding
DOC	Direct Operating Cost
DS	Directionally Solidified
EPNdB	Effective Perceived Noise Decibel
EW	Empty Weight
FAR	Federal Aviation Regulation
FH	Flight Hour
F_N	Net Thrust
FOD	Foreign Object Damage
F/W	Specific Thrust, (Thrust per Unit Airflow)
HP	Horsepower/High Pressure
HPC	High-Pressure Compressor
HPT	High-Pressure Turbine
IFR	Instrument Flight Rules
IOC	Indirect Operating Cost
ISA	International Standard Atmosphere
KEAS	Knots Equivalent Air Speed
KTAS	Knots True Air Speed
LART	Low Aspect Ratio Turbine
LP	Low Pressure
LPC	Low-Pressure Compressor
LPT	Low-Pressure Turbine
M_n	Mach
MCR	Maximum Cruise Thrust
MCT	Maximum Climb Thrust
NASA	National Aeronautics and Space Administration

NMI	Nautical Mile
OASPL	Overall Sound Pressure Level
OCR	Overall Compression Ratio
OEI	One Engine Inoperative
OEM	Original Equipment Manufacturer
PAX	Passengers
PLA	Power Level
PM	Powder Metal
P/P	Pressure Ratio
PSIA	Pounds Per Square Inch Absolute
QEC	Quick Engine Change
SC	Single Crystal
SFC	Specific Fuel Consumption
SHP	Shaft Horsepower
SI	International System of Units
SLS	Sea Level Static
SPF	Super Plastic Forming
SSM	Seat Statute Mile
STAT	Small Transport Aircraft Technology
TBO	Time Between Overhaul
TO	Take-Off
TOGW	Take-Off Gross Weight
TRIT	Turbine Rotor Inlet Temperature
TSFC	Thrust Specific Fuel Consumption
WA	Airflow Rate
WAF	Airframe Weight
η	Efficiency
η_B	Baseline Efficiency
ΔH	Enthalpy Change
ΔH_B	Baseline Enthalpy Change

APPENDIX III

STAT ADVANCED ENGINE CYCLE AND CONFIGURATION DEFINITION AND OFF-DESIGN PERFORMANCE

APPENDIX III

Table XXVIII defines the cycle and configuration or the 1990 Advanced Technology Engine. The characteristics shown are for an installed engine in the baseline size (10 lb/sec inlet corrected flow) the engine is comprised of a two-stage centrifugal compressor producing a 16:1 pressure ratio, a reverse flow annular combustor, a two stage cooled axial high-pressure turbine and a two stage uncooled axial low-pressure turbine. Cycle assumption not shown in Table I included:

-	Aircraft accessory horsepower	- 15 HP
-	Aircraft bleed extraction	- 12 lb/min
-	Leakage	- 1 1/2 compressor exit airflow
-	Engine mechanical losses	- 10 HP + 0.5% of compressor HP
-	Interturbine % P/P	- 2
-	LP turbine exit losses % P/P	- 1.5%
-	Nozzle thrust coefficient	- 0.93

Table XXIX shows flight envelope performance for the engine. Performance is shown at the engine design point - 17,000 feet, 291 knots cruise power-and at sea level, 90°F, static conditions.

Performance is also shown for a range of altitudes, flight velocities and power settings. Output power and fuel consumption are based on gearbox output. Net jet thrust (F_N) is the contribution of the exhaust nozzle.

Figure 40 shows the effect of bleed air and accessory horsepower extraction on the 1990 technology engine at three operating conditions. Shown are the changes in shaft horsepower and specific fuel consumption as bleed airflow and/or accessory horsepower loads are decreased from their nominal values (W_B = 12 lb per min; HP_{acc} = 15) to zero.

TABLE XXVIII

APPENDIX III
CYCLE AND CONFIGURATION DEFINITION
STAT ADVANCED ENGINE
INSTALLED PERFORMANCE
[(10 lb/sec) Core Size]

Altitude, M(FT)	0	17,000
Mach No.	0	0.468
Ambient Temperature °C (°F)	15 (59)	-18.7 (-1.6)
Power Setting	TO	MXCR
Cycle Pressure Ratio	16.1	16.0
Turbine Inlet Temperature °C (°F)	1371 (2500)	1288 (2350)
Output Power Kw (HP)	1679 (2252)	1039 (1394)
Brake Specific Fuel Consumption KG/w-h (LBM/HP-H)	0.253 (0.416)	0.239 (0.393)
Net Jet Thrust (N (LB)	(192)	(32)
Inlet Corrected Flow KG/S (Lb/Sec)	(10.1)	(10.0)
Inlet Flow KG/S (Lb/Sec)	(10.1)	(6.27)
<u>Compressor</u>		
No of Stages	2	
Efficiency	0.831	0.831
<u>HP Turbine</u>		
No of Stages	2	2
Efficiency	0.897	0.895
Corrected Specific Work (BTU/LB)	(34.36)	(34.26)
Pressure Ratio	3.432	3.436
Chargeable Cooling Flow % W_A	5.7	5.7
<u>LP Turbine</u>		
No of Stages	2	2
Efficiency	0.890	0.892
Corrected Specific Work (BTU/LB)	(37.76	(40.28)
Pressure Ratio	3.999	4.477
Chargeable Cooling Flow % W_A	1.0	1.0
<u>Combustor</u>		
Pressure Drop, ΔP/P	0.051	0.05
Combustion Efficiency	0.995	0.995
Exhaust Nozzle Pressure Ratio	1.081	1.10
<u>Dimensions</u>		
Engine Length - M (IN)	1.334(52.5)	1.334(52.5)
Engine Max Diameter - M (IN)	0.787(31.0)	0.787(31.0)
Engine Weight - KG(LB)	282.6 (623)	282.6 (623)

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TABLE XXIX

APPENDIX III

STAT 1990 ENGINE

Altitude	Mach	PLA	SHP	SFC	FN
17,000	0.4681	MCR	1404	0.391	32.3
SL 90°F	0	TO	1902	0.437	158.1
SL 59°F	0	TO	2252	0.417	192.3
			1726	0.439	147.5
			1235	0.482	109.6
			783	0.573	77.1
			[illegible]90	0.824	50.4
	0.2	TO	2288	0.413	120.2
		MCT	2087	0.419	104.9
		MCR	1885	0.428	90.5
			1374	0.464	57.8
			907	0.536	31.7
			491	0.718	12.4
			195	1.292	1.6
	0.4	TO	2447	0.401	60.4
		MCT	2230	0.407	46.1
		MCR	2016	0.414	33.1
			1482	0.445	5.3
			988	0.509	-14.7
			558	0.657	-26.7
			254	1.045	-31.8
	0.6	TO	2704	0.385	0
		MCT	2467	0.389	-13.9
		MCR	2239	0.394	-25.9
			1666	0.418	-50.3
			1124	0.470	-65.0
			674	0.578	-70.9
			357	0.804	-71.2
	0.8	TO	3061	0.366	-64.5
		MCT	2807	0.368	-78.6
		MCR	2558	0.371	-90.4
			1914	0.389	-112.9
			1325	0.427	-122.8
			860	0.493	-124.4
			510	0.618	-120.8

TABLE XXIX (Contd)

APPENDIX III

STAT 1990 ENGINE

Altitude	Mach	PLA	SHP	SFC	FN
10,000 ISA	0.2	TO	1770	0.408	105
		MCT	1634	0.412	93
		MCR	1490	0.418	82
			1099	0.448	54
			743	0.508	32
			412	0.662	15
			164	1.156	4
	0.4	TO	1904	0.397	64
		MCT	1754	0.400	52
		MCR	1597	0.405	41
			1186	0.431	16
			809	0.484	-2
			465	0.611	-15
			211	0.956	-21
	0.6	TO	2118	0.381	23
		MCT	1946	0.384	11
		MCR	1772	0.388	-1
			1331	0.407	-24
			919	0.450	-40
			555	0.545	-47
			293	0.749	-49
	0.8	TO	2398	0.365	-19
		MCT	2207	0.365	-33
		MCR	2019	0.367	-45
			1539	0.380	-68
			1078	0.412	-82
			688	0.476	-85
			413	0.588	-85
20,000 ISA	0.2	TO	1316	0.404	84
		MCT	1235	0.406	76
		MCR	1146	0.408	69
			878	0.429	47
			608	0.477	29
			360	0.589	15
			147	0.980	4
	0.4	TO	1428	0.393	58
		MCT	1337	0.394	50
		MCR	1237	0.396	41

TABLE XXIX (Contd)

APPENDIX III

STAT 1990 ENGINE

Altitude	Mach	PLA	SHP	SFC	FN
20,000 ISA	0.4		946	0.414	21
			663	0.455	5
			401	0.550	-7
			184	0.830	-13
	0.6	TO	1605	0.379	34
		MCT	1499	0.378	24
		MCR	1384	0.380	15
			1059	0.394	-7
			757	0.424	-21
			472	0.499	-30
			247	0.675	-33
	0.8	TO	1830	0.366	12
		MCT	1707	0.364	-1
		MCR	1576	0.363	-13
			1221	0.370	-36
			885	0.392	-50
			573	0.445	-57
			338	0.547	-57
30,000 ISA	0.2	TO	908	0.407	61
		MCT	865	0.407	57
		MCR	816	0.407	52
			652	0.420	38
			460	0.460	24
			281	0.554	13
			111	0.923	4
	0.4	TO	994	0.396	45
		MCT	945	0.395	41
		MCR	889	0.395	35
			707	0.406	21
			503	0.440	7
			314	0.519	-2
			139	0.785	-8
	0.6	TO	1131	0.383	34
		MCT	1072	0.380	27
		MCR	1006	0.379	20
			796	0.387	3
			575	0.412	-10

TABLE XXIX (Contd)

APPENDIX III

STAT 1990 ENGINE

Altitude	Mach	PLA	SHP	SFC	FN
30,000	0.6		369	0.473	-18
			186	0.644	-21
	0.8	TO	1321	0.368	25
		MCT	1239	0.367	16
		MCR	1155	0.366	7
			918	0.366	-15
			678	0.382	-28
			447	0.426	-36
			255	0.527	-37
40,000	0.2	TO	559	0.419	37
ISA		MCT	532	0.419	35
		MCR	500	0.420	31
			392	0.438	22
			261	0.495	13
			132	0.672	5
			18	2.791	0
	0.4	TO	616	0.406	28
		MCT	585	0.405	25
		MCR	549	0.406	22
			430	0.421	12
			292	0.468	3
			159	0.601	-3
			43	1.341	-6
	0.6	TO	707	0.392	21
		MCT	671	0.389	17
		MCR	629	0.389	13
			492	0.399	2
			344	0.433	-7
			201	0.524	-12
			84	0.828	-14
	0.8	TO	837	0.375	17
		MCT	786	0.374	11
		MCR	731	0.374	6
			578	0.376	-9
			417	0.396	-18
			259	0.456	-23
			136	0.603	-23

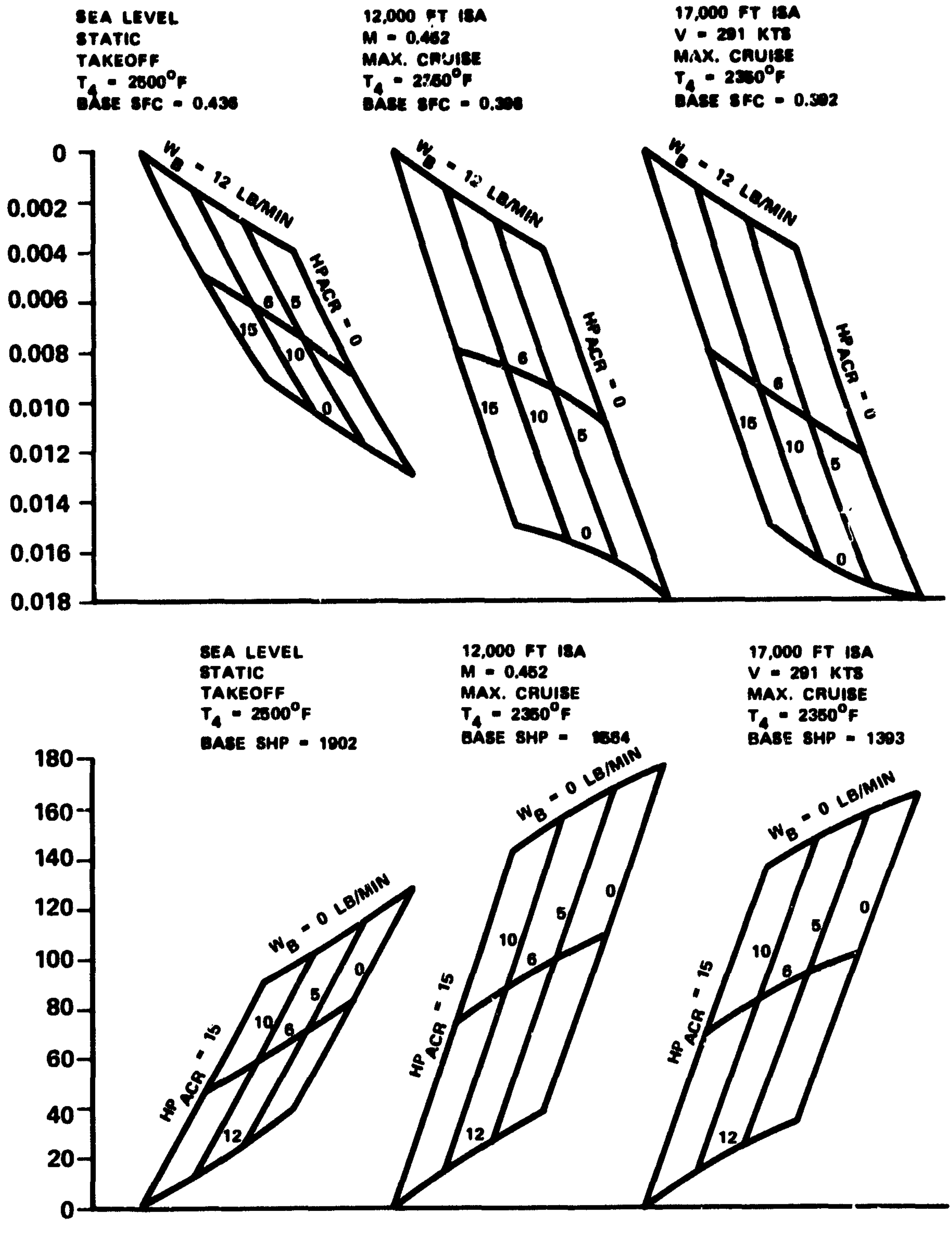

Figure 40. STAT 1990 Technology Engine Bleed-Air and Accessory Load Effects.

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