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SUMMARY

This analysis explores the benefits forecast for advanced turboprop and turboshaft engines in the expanding general aviation fixed- and rotary-wing market. Although turboshaft engines are currently used in this class of helicopters, turboprops have been unable to penetrate the light fixed-wing market because of their high acquisition cost (three times that of a current spark ignition reciprocating (SIR) engine) and their specific fuel consumption (25 percent higher than current SIR). Advanced technology and new production techniques may improve this situation.

Compared with a current production turboprop, an advanced technology (1988) turboprop results in a 23 percent decrease in specific fuel consumption (ESFC). The same advanced engine when compared with a hypothetical engine using currently available technology (1978) results in an 8 percent improvement in ESFC and a 22 percent decrease in engine weight.

The present study determines the effect of these improvements on such figures of merit as vehicle gross weight, mission fuel, airplane acquisition cost, operating cost, and life cycle cost for three fixed-and two rotary-wing aircraft.

For a light twin airplane, an advanced technology turboprop uses 20 percent less fuel than a current SIR engine and is competitive with a hypothetical advanced SIR engine (10 percent lower BSFC, 33 percent lower weight/horsepower).

The optimum (based on minimum operating cost) advanced turboprop for this airplane has a three axial plus one-centrifugal stage compressor, a two-stage axial high pressure turbine (HPT), and a two-stage axial low pressure turbine (LPT). The engine has a design point pressure ratio at cruise of 12 and a turbine rotor-inlet temperature (TIT) of 2600° R. This is also the optimum engine for the medium twin.

For the single-engine aircraft, the optimum advanced turboprop is based on minimizing airplane acquisition cost. As a result engine cost is of major importance. Therefore, this engine has a single-stage centrifugal compressor, a radial HPT and a two-stage axial LPT. The cruise design point compressor pressure ratio is 9 and the TIT is again, 2600° R.

To compete economically with a SIR aircraft, the cost of an advanced turboprop will (depending on the figure of merit and the mission) have to be reduced considerably. As a powerplant for a six-place medium twin, an advanced turboprop is better than a current production SIR engine and competitive with a hypothetical advanced SIR engine in terms of all of the figures of merit except acquisition cost. To achieve the same airplane acquisition cost as the SIR engines requires turbine engine cost ($/SHP-OEM) reductions of 52 and 67 percent, respectively. For smaller aircraft, such as the six-place single engine aircraft, the turboprop is less competitive. For this aircraft the needed turbine engine cost reductions increase to 60

and 74 percent. Based on life cycle cost (5 years) for the single-engine aircraft, slightly lower engine cost reductions (42 and 65 percent) are required to achieve cost parity with SIR engines. According to various engine manufacturers, cost reductions of this magnitude maybe achieved by means of advanced technology and high production.

Similarly an advanced technology turboshaft results in significant improvements relative to a hypothetical engine using currently available technology. These improvements include an 11 percent reduction in vehicle acquisition cost, a 16.9 percent reduction in mission fuel and an 11.4 percent improvement in life cycle cost when powering a light twin-engine helicopter. Based on minimum operating cost, the optimum advanced turboshaft has the same configuration and cycle for both a light single and a light twin helicopter. This configuration consists of a two stage centrifugal compressor, a radial HPT, and a two stage axial LPT. The engine has a sea level static design point compressor pressure ratio of 12 and a TIT of 2760° R.

SYMBOLS

A	axial stage; area
BPR	bypass ratio
BSFC	brake specific fuel consumption, lb/hr - HP
C	centrifugal stage
C_D	airplane drag coefficient
C_{Do}	airplane zero-lift drag coefficient
CF	correction factor
ESFC	equivalent specific fuel consumption, lb/hr - HP
ESHP	equivalent shaft horsepower
g	gravitational constant, ft/sec^2
HP	horsepower
HPT	high pressure turbine
J	conversion factor ft-lb/BTU
LCC	5-year cost of ownership
LPT	low pressure turbine
OEM	original equipment manufacturer
OPR	compressor overall pressure ratio

P	pressure lb/in^2
P_2	ambient sea level pressure, lb/in^2
Pr	pressure ratio
RPM	revolutions per minute
SIR	spark ignition reciprocating
T	temperature, ^{o}R
T_2	ambient sea level temperature, ^{o}R
TBO	time between overhaul, hr
T_c	compressor exit bleed temperature, ^{o}R
T_m	bulk metal temperature, ^{o}R
Tg	turbine inlet gas temperature, ^{o}R
TAS	true airspeed, kts
TIT	turbine rotor-inlet temperature, ^{o}R
TNP	total number produced
TOGW	takeoff gross weight, lb
TP	turboprop
TSFC	thrust specific fuel consumption, lb/lb-hr
Um	mean blade velocity, ft/sec
Ut	tip velocity, ft/sec
Wa	airflow, lb/min
W_E	aircraft empty weight
W_{comp}	compressor airflow, lb/sec
W_{cool}	coolant airflow, lb/sec
δ	corrected pressure
θ	corrected temperature
ΔH	enthalpy change per stage, BTU/lb
η	adiabatic efficiency
η_p	polytropic efficiency

Subscripts:

d design

SL sea level

INTRODUCTION

Today almost all segments of aviation are turbine powered with the exception of general aviation light airplanes. These airplanes, requiring 100-450 horsepower, are for the most part powered by spark ignition reciprocating (SIR) engines. The inability of the turbine to penetrate the general aviation light airplane market is due mainly to its high acquisition cost (approximately three times that of a SIR engine) and high equivalent specific fuel consumption (25 percent higher (ref. 1)).

If these two obstacles can be overcome, the many advantages of the turbine relative to the SIR engine may be realized. From a passenger viewpoint, these advantages are less vibration, higher reliability, greater safety, and less noise. From an owner's viewpoint, the turbine possesses a multi-fuel capability (important in light of the energy crisis) and requires fewer overhauls as indicated by its greater time between overhauls (TBO - 1500 versus 3000 hours). In addition, the turbine engine weighs one-third that of a SIR engine and has lower installation and drag losses.

Some current production turbine engines in the 400-700 horsepower category are listed in Table 1. The engines have overall pressure ratios of about 8, turbine rotor-inlet temperature of 2300° R, weigh about 0.52 lbs/hp (based on dry weight), and have an ESFC of about 0.6 lb/hr/hp. Most of these engines were certified in the 60's or 70's and are based on technologies which existed 10-20 years ago.

To explore new opportunities for turbine engines in the expanding general aviation market, NASA/Lewis Research Center initiated four contracted studies (refs. 2 to 5) and an in-house study, which is the subject of this report.

The present study explores the benefits forecast for advanced turboprop and turboshaft engines by way of advanced technology. To maximize these benefits various engine parameters such as configuration, compressor overall pressure ratio (OPR), and turbine rotor-inlet temperature are varied to search for the optimum design. Three fixed-and two rotary-wing aircraft are considered for these engines. The vehicle figures of merit used to evaluate these parametric engines were vehicle gross weight (TOGW), mission fuel, acquisition cost, operating cost, and life cycle cost (LCC). In addition to turboprop and turboshaft engines, turbofans are reviewed.

The variations in engine configurations were limited to changes in the type of compressors and gas generator turbines used. Compressor configurations included both one- and two-stage centrifugals and a number of axial-centrifugal arrangements. Overall pressure ratios for these compressors ranged from 8 to 16. The gas generator turbine configurations evaluated were a one-stage radial, one- and two-stage axial, and a radial-axial arrangement. Turbine rotor-inlet temperatures (TIT) ranged from 2200° to 2950° R. The advanced technology turboprop and turboshaft engines are compared with current technology turbine engines as well as current and advanced SIR engines. The cost reduction required for the advanced turboprop to achieve aircraft cost parity with the SIR engines is determined along with the sensitivity of aircraft acquisition cost, operating cost, life cycle cost, and mission fuel to component efficiencies.

ANALYSIS

Missions

The three fixed-and two rotary-wing aircraft categories considered in the present study are shown in Table II.

The three fixed-wing categories consist of a high performance single engine, a light twin, and a medium twin. All three are six-place airplanes and are representative of current SIR airplanes. The assumed cruise speeds are 174 kts for the single engine, 226 for the light twin, and 234 for the medium twin; the corresponding cruise altitudes are 7500, 10 000, and 25 000 feet, respectively. Only the medium twin is pressurized.

The two rotary-wing categories consist of a light single and a light twin engine aircraft. The former is a four-place while the latter is a six-place aircraft. Cruise speeds vary from 110 kts for the light single to 125 kts for the light twin. The hover ceiling altitudes are 6000 feet for the single and 8000 feet for the twin. Ranges are 300 N.mi for the single and 450 N.mi for the twin.

Takeoff gross weight (TOGW), mission fuel, airplane acquisition cost, operating cost, and life cycle cost are used as the figures of merit.

The General Aviation Synthesis Program (GASP) (ref. 6) is used to size the fixed-wing, current technology aircraft for the required payload and range, determine the drag buildup, establish the power requirements, and calculate the acquisition and operating costs. With a SIR engine, the calculated airplane drag (CD) is increased by 11 percent to account for cooling drag. The operating cost equations of reference 7 are used in place of similar equations in GASP. Fuel costs are assumed to be $1/gallon for the turbine and $1.10/gallon for the SIR. Life cycle costs (LCC) are based on 5 years of ownership and are calculated by adding the acquisition, fuel, maintenance, overhaul, insurance, storage, FAA tax, and interest costs while subtracting the trade-in allowance (70 percent of the acquisition cost).

A computer code similar to the one given in reference 8 is used to size and weigh the rotary-wing aircraft for the required payload and range, establish the power requirements and calculate the acquisition, operating, and life cycle costs. Acquisition and life cycle costs are based on the same ground rules as used in the fixed-wing analysis.

Engines

Turbomachinery efficiencies. - Two levels of compressor and turbine efficiencies are considered for the hypothetical engines considered in the current study. One level represents current technology (1978) as opposed to the second level which represents advanced (1988) technology (see Appendix C). Compressor efficiency varies with stage pressure ratio and compressor type (axial or centrifugal) while turbine efficiency varies with turbine stage work factor (g $J\Delta H/Um^2$ (see Appendix C for definition)) and turbine type (axial or radial). In addition to correcting the compressor and turbine efficiencies for size effects, turbine efficiency is also corrected for tip clearance and cooling effects.

Cooling. - Turbine cooling requirements are based on a procedure similar to the one given in reference 9. Values for cooling effectiveness for the advanced technology engines are based on full-coverage film-cooling of reference 10. Cooling effectiveness is defined as the ratio of the difference between the hot gas temperature (Tg) and the allowable bulk metal temperature (Tm) to the difference between the hot gas temperature and the compressor bleed temperature (Tc). The hot gas temperature is defined as the average combustor exhaust temperature incremented to include the effects of (1) hotspot profiles, (2) dilution due to the cooling air injection from upstream rows, (3) relative velocity, (4) work extraction, and (5) a safety margin of 150° R. The procedure of reference 9 did not allow for either a safety margin or a hotspot temperature. The cooling requirements are based on a row-by-row calculation procedure using compressor exit bleed air and a calculated value of cooling effectiveness. Based on advanced technology, the allowable bulk metal temperature for a vane is projected to be 2240° R.

For a rotor blade, the allowable bulk metal temperature is 100° R lower because of the higher stresses. The decrease in turbine efficiency due to turbine cooling is discussed in Appendix C.

Power and bleed extraction. - For turbine engines, the aircraft auxiliary horsepower and bleed extraction (for cabin pressurization) requirements vary with the mission. A power extraction of 8 horsepower per engine is assumed for the twins and 5 horsepower per engine for the single-engine and both rotary-wing aircraft. A bleed flow equal to 3 percent of the compressor discharge flow per engine is assumed for pressurizing the medium fixed-wing twin. No bleed is assumed for the other aircraft since they are all unpressurized.

Engine performance. - Turbine engine performance was computed from forecasted design point component performance trends and approximate off-design engine performance.

Some important engine parameters considered in the study and their effect on design point performance for small turbine engines ($\leq$2 lbs/sec) are illustrated in Figure 1. The figure illustrates the effects of (1) variation in turbomachinery efficiency with pressure ratio, (2) turbine cooling, and (3) engine size (correction of efficiency for size) on performance and cycle selection. As indicated, the degree of realism included in small turbine engine cycle calculations can have a significant effect. The performance illustrated in part A at the [illegible] the figure assumes constant compressor and turbine efficiency, no tu[illegible] cooling and a large engine (no size correction). For a TIT of 2900° F, BSFC decreases with increasing compressor pressure ratio (CPR). The BSFC for a CPR of 50 is 0.312. Decreasing the CPR from 50 to 14 for a TIT of 2900° F causes the BSFC to increase 26 percent.

Part B indicates the importance of varying the compressor efficiency with pressure ratio and the turbine efficiency with stage work factor (Appendix C). Now for a TIT of 2900° F, the optimum CPR is 50 and the BSFC is 0.35 which is 12 percent higher than for Part A. Now decreasing the CPR to 14 results in only a 14 percent increase in BSFC. Thus, the lower turbomachinery efficiencies reduce the benefits of high cycle pressure ratio.

Next, in part C, turbine cooling requirements (TIT > 1900° F) are taken into account. The dotted section is uncooled and repeats a portion of Part B. ere the turbine blades do not have to be structured to incorporate cooling passages and there is no decrement in efficiency. In the slashed portion, turbine efficiency is penalized for thicker blades (required to incorporate cooling passages), and the cycle performance is penalized due to the effect of cooling flow requirements. As a result, the optimum pressure ratio and TIT are reduced to 18 and 2700° F. The resulting BSFC is 0.43.

Lastly, in part D the turbomachinery efficiencies are reduced to account for the reduction in engine inlet corrected airflow from 10 to 2 lbs/sec. Now the optimum compressor pressure ratio and TIT are about 12 and 2900° F, and the BSFC is 0.525. Thus improving the realism of the cycle calculation by accounting for (1) variations in turbomachinery efficiency, (2) turbine cooling effects, and (3) size effects limits the benefits of high cycle pressure ratio and high turbine rotor-inlet temperature, thereby resulting in significantly higher BSFC's (68% for the example case of Figure 1). Hence due to their importance, these effects are incorporated in all of the turbine data of this study.

Spark ignition reciprocating (SIR) engine performance is an integral part of the GASP program. The program uses generalized, nondimensional relationships between (1) power and rpm and (2) power and altitude to predict the full power of an engine at any combination of rpm and altitude. Fuel flow is a function of engine displacement and percent rated power. A turbocharged engine is assumed to maintain its rated power up to some critical altitude (16 000 feet was assumed for the medium twin) above which power decreases with altitude.

Weight and dimensions. - Turbine engine dimensions and core weight are calculated by the procedure of reference 11. Included in the weight of the core engine are the effects of bypass ratio (BPR), overall pressure ratio (OPR), turbine rotor-inlet temperature (TIT), airflow, and technology level. The relationships can be used to calculate the weight of a small turbofan or turbojet, but not a turboshaft engine. Based on reference 12, the weight of a turboshaft engine is determined by assuming it is a turbojet and increasing the resultant weight by a factor of 3. This weight penalty is due to the large power turbine, the extra shaft and bearing, and a larger and stronger case needed for the larger structural loads. The calculated weights were found to be in good agreement with existing turboshaft engines. In addition, a turboprop engine includes a gearbox. The gearbox weight is calculated in GASP. For an advanced (1988) technology), 300 HP_{SLS} turboprop, having a cruise design point TIT of 2600° R, and an OPR of 12, the uninstalled specific weight is calculated to be 0.51 lbs/HP. The weight of the same engine is 22 percent heavier when based on current technology. Engine installation weight is determined in the GASP program. This weight includes the following: air induction system, lubrication system, starting system, engine controls, and nacelle.

The weight and dimensions for the SIR engines are calculated in GASP. Typical values of specific weight for current naturally aspirated and turbocharged engines are 1.52 and 1.66 lbs/HP, respectively, for approximately 400 HP engines.

Propeller

Propeller data used in the study is indicated below:

Engine	SIR	TP
Propeller Efficiency		
Current (1978)	0.87	0.89
Advanced (1988)	0.89	0.91

Airplane	Single engine	Light twin	Medium twin
Propeller Data			
Diameter, in.	84.6	91	91
No. of blades	2	3	3
Weight	Reference 13		

The efficiency of turboprop propellers is assumed to be higher than SIR engine propellers due to the lower vibrational stresses associated with a turboprop.

Engine cost. - Engine cost (original equipment manufacturer - OEM) is described in Appendix A.

RESULTS AND DISCUSSION

Engine Comparison

Table III compares the smallest current production turboprop engine with a hypothetical engine of the same shaft horsepower using currently available technology. The equivalent specific fuel consumption (ESFC) of the hypothetical engine is 15 percent less than the current production engine. Since it was not know what was included in the published weight of the production engine, its weight was calculated with the turbine engine weight routine used in this study. Compared with the calculated current production uninstalled engine weight, the hypothetical engine is 7 percent lighter.

For the hypothetical engine, the pressure ratio (at the cruise design point) was increased from 7.2 to 12 and the TIT from 2457° to 2600° R. A pressure ratio of 12 is achieved with three axial stages (P/P = 1.3 per stage) and one centrifugal (P/P = 5.46) stage. The high pressure and low pressure turbines each have two axial stages. Turbine cooling is based on convection cooling with trailing edge ejection. A 2 percent blade tip clearance is assumed for the turbines. Turbomachinery performance is based on the current technology curves discussed in Appendix C.

Current technology TP versus current SIR. - Engine characteristics are compared in table IV for two propulsion systems. These are a turbocharged SIR engine and a hypothetical current technology TP, as installed in a "rubberized" six-passenger medium twin. For the turbocharged SIR engine, performance (BSFC = 0.45), weight (1.66 lb/SHP), and cost (32.4 $/SHP - OEM) are characteristic of current production turbocharged engines. For turbocharged SIR engines, current critical altitudes vary from about 12 000 to 20 000 feet depending on the engine. For this study, a value of 16 000 feet was selected. Cooling drag for a SIR engine has been estimated to account for between 5 and 20 percent of the total cruise drag of the airplane (ref. 14). For this study, the airplane cruise drag was increased by 11 percent to account for engine cooling drag. Thus, the critical altitude and the cooling drag penalty for the SIR engine could be considered to be either optimistic or pessimistic depending on one's point of view. The hypothetical current technology turboprop is a scaled-down version of the turboprop described in the previous paragraph. The main disadvantage of this turboprop relative to the turbocharged SIR engine is its current high cost (2.75 times the turbocharged SIR). Turboprop costs used in this study are discussed in Appendix A.

Compared with the turbocharged SIR engine, the ESFC disadvantage for the hypothetical current technology turboprop is only 3 percent. However, when the same turbocharged SIR engine is compared with a current production TP, the ESFC disadvantage for the turboprop is of the order of 18 percent (based on the previous section). In terms of engine weight, installation losses, TBO, and multi-fuel capability, the hypothetical current technology turboprop is superior to the SIR engine. The turboprop weighs about one-third that of a SIR engine, has zero cooling drag and 1800 additional hours before an overhaul. Based on this study, turbine engine maintenance and overhaul costs are competitive with SIR engines. In terms of the overhaul cost, the higher cost of the turboprop is offset by its greater TBO. As indicated, maintenance and overhaul costs used in this study are based on current production turbine engines. Thus for the current hypothetical engine and especially the advanced technology turboprop these costs need to be reviewed.

Mission results for the hypothetical turboprop and the turbocharged SIR engines are indicated in Table V. Both the turboprop and the SIR engine are sized at the cruise design point. For the SIR this operating point corresponds to 60 percent of rated power at 90 percent of maximum RPM. Because of the turboprop lapse rate, the cruise operating point corresponds to 57 percent of rated power. The turboprop requires a lower horsepower engine due to the lower TOGW (6002 vs. 7841 lb) resulting fom the lower engine weight (410 vs. 1417 lb), and installation losses. As a result, the turboprop uses 26 percent less fuel even though its ESFC is 3 percent greater. The higher engine cost associated with the turboprop is somewhat offset by the lower airframe cost (due to lower airframe weight). However, the total airplane acquisition cost is $88 000 higher for the turboprop aircraft than for the turbocharged SIR engine. In terms of operating cost (based on a utilization rate of 800 hrs/yr), the reverse is true; i.e., the turboprop cost is lower. Most of this difference is due to the lower fuel cost (less fuel and lower cost per gallon). Jet-A is assumed to cost 10 percent less than Av gas. Therefore, as a propulsion system for a medium pressurized twin, a hypothetical turboprop using currently available technology appears to be competitive with a current production turbocharged SIR engine. However, this picture changes considerably if one reconsiders the critical altitude and cooling drag for the turbocharged SIR engine.

Values used for the critical altitude and cooling drag of the turbocharged SIR engine have a significant effect on the numbers quoted in the previous paragraph. For example, increasing the critical altitude from 16 000 to 20 000 feet, and decreasing the cooling drag from 11 to 0 percent results in a 23 percent decrease in the mission fuel for the turbocharged SIR engine (fig. 2). With these changes in ground rules, the hypothetical current technology turboprop now uses only 4 percent less fuel than the turbocharged SIR engine. Thus, differences in these two parameters can produce significant differences in results between studies.

Advanced turboprop technology - Compared with the hypothetical, current-technology turboprop discussed in the previous section (which is more advanced than current production engines), an advanced 1988 turboprop will incorporate additional component improvements projected for this time period. These are discussed in detail in appendix C and are based on existing data and discussions with NASA component specialists. Component efficiencies are increased as follows:

Axial stage compressor	$\sim$1.5 percent
Centrifugal stage compressor	2.0 percent
Axial high-pressure turbine (HPT)	0.6 percent (zero tip clearance)
Radial HPT	1.7 percent
Axial low-pressure turbine (LPT)	.4 - 1.6 percent

In addition pressure ratios as high as 12 are considered for a single-stage centrifugal compressor. These compressors will employ three-dimensional blading and be fabricated to essentially net shape from powder metal to reduce cost. The maximum tip speed for the radial turbine is increased from 1800 to 1900 ft/sec., thereby, improving performance. Tip clearances for the axial turbine are decreased from 2 to 1 percent of blade height resulting in an additional 2 percent increase in efficiency. For the same turbine rotor-inlet temperature (TIT) and blade metal temperature, the turbine cooling flow requirements are decreased 30 percent by means of full film-cooling. A further decrease in the cooling requirement is achieved by means of a 100° R increase in blade metal temperature (directionally solidified superalloy). An advanced 1988 turbine engine will also weight about 20 percent less. For the turboprop gearbox, laser contour hardening of the gear teeth will be developed to improve gear life and reduce cost. Digital electronic controls will also be used for these advanced general aviation turbine engines to achieve low cost and high reliability. To achieve all of these improvements will be a challenge.

Turboprop versus turbofan. - To determine if an advanced turbofan would be an attractive alternative propulsion system for this category of airplane, a brief study was undertaken. The results are indicated in Table VI. Both engines use the same core cycle, configuration, advanced turbomachinery technology and are sized to provide 318 pounds of thrust at 234 kts and 25 000 feet. The fan pressure ratio and BPR of the turbofan were varied to minimize TSFC. The resultant cruise TSFC for the turbofan is 56 percent higher than for the turboprop. Furthermore, the core airflow for the turbofan is 86 percent higher than for the turboprop when both are sized for the same cruise thrust. As a result, the turbofan core will cost more due to its larger size. Therefore, the turbofan is not considered to be as attractive as the turboprop for use in general aviation light aircraft.

Thus, only the turboprop is considered for the fixed-wing aircraft in the remainder of this study.

Cycle and Staging Arrangement Optimization

For the fixed- and rotary-wing missions under study, one would like a turboprop and a turboshaft engine having good performance (low ESFC) at a low cost. Unfortunately, the two do not go hand-in-hand. In these applications, engine cost is of paramount importance, so much so that engine performance may be traded to reduce cost. To some degree, this trade is considered in this study. However, the engine cost models utilized were not sensitive to changes in several important variables: Engine cost is affected by compressor overall pressure ratio (OPR), but not by part count. Therefore, on a cost basis, the choice between a single and a multi-stage compressor having the same OPR is not clear. Part count is taken into account in determining the cost of the high pressure turbine (HPT) but not the low pressure turbine (LPT). Like the compressor, engine cost is not affected by the type of turbine, radial or axial. Engine cost is affected by TIT. To overcome these deficiencies, certain components are treated on a parametric basis. With this as a background, the next two sections determine the optimum engine configuration and cycle for the fixed-wing light twin and the single-engine light helicopter on the basis of various figures of merit. A similar procedure was followed for the other missions.

Fixed-Wing Engine Optimization

Compressor. - Figure 3 indicates the effects of compressor type and OPR on TOGW, mission fuel, acquisition cost, and operating cost (800 hrs/yr). As noted in the figure, the turbine configuration and the TIT are fixed. With a single-stage centrifugal compressor, the optimum pressure ratio is between 9 and 10.5 for all of the figures of merit. With a two-stage centrifugal (60-40 Pr split) or three-axial (P/P - 1.3/stage) plus one-centrifugal compressor, the optimum pressure ratio is between 10 and 15 (depending on the figure of merit). Three axial stages were found to be optimum. Reduced fuel usage favors higher pressure ratios due to lower ESFC while acquisition cost favors lower pressure ratios due to lower engine cost.

The multi-stage compressors are better (based on the figures of merit) than the one-stage centrifugal due to their higher efficiency at a given pressure ratio. Acquisition costs are reduced because of the decrease in airframe and engine costs (assuming cost is independent of part cost). Operating costs are reduced because of the decrease in mission fuel and aircraft related costs. An axial-centrifugal compressor having a pressure ratio of 12 was selected as the optimum for the light twin, based on minimum operating cost.

Turbine rotor-inlet temperature. - The three-axial plus one-centrifugal stage compressor configuration (OPR = 12) was used to determine the effect of turbine rotor-inlet temperature (TIT) on the various figures of merit for the light twin. Based on all of the figure of merit, a TIT of about 2600° R results in a superior airplane (fig. 4) and, therefore, was selected as the design point TIT for the remainder of the fixed-wing portion of the study. Although higher temperatures appear to be advantageous, they are considered to be beyond the technology assumed for the 1988 time frame. References 2 to 5 also considered approximately the same temperature to be an upper limit for the missions they considered. For larger engines (Wa $\sqrt{\theta}/\delta \geq 100$ lb/sec) increasing TIT tends to improve engine performance (lower ESFC) and increase engine specific thrust, thereby reducing engine weight and cost. As previously noted (fig. 1), component size effects temper these improvements for the small engines being studied.

HPT configuration. - A two-stage axial and a one-stage radial HPT were studied for driving the axial-centrifugal compressor configuration, figure 5. The radial turbine results in a vehicle having a slightly lower TOGW as a result of its slightly better efficiency and, therefore, better fuel economy. Since the engine cost model considered the number of stages in the HPT (see Appendix A), the one-stage radial turbine also results in a vehicle having a lower acquisition cost. The combination of less fuel and lower acquisition cost results in the radial turbine having a slightly lower operating cost (1.2 percent). However, a more detailed engine cost study is required to substantiate these cost results. Based on these preliminary results, both configurations are attractive especially when the optimum configuration is based on minimizing aircraft operating cost. The two-stage axial HPT configuration was selected for the optimum engine (fig. 6) mainly because of the greater experience and, thus, possibly lower risk of an axial stage turbine.

Rotary-Wing Engine Optimization

Studies similar to those for the turboprop were also made for the turboshaft engine to determine the optimum engine configuration and cycle for a single and a twin-engine light helicopter (defined in table II). The study results for the single-engine, light helicopter are presented in the next two figures. The engine design point was the sea level static condition.

Compressor. - Four compressor configurations were considered for the single-engine light helicopter, figure 7. These included a one-stage centrifugal, a two-stage centrifugal (60-40 pressure ratio split), and two axial-centrifugal configurations. One axial-centrifugal configuration had the pressure ratio for the centrifugal stage fixed at 2.5 (based on several production engines); the number of axial stages (P/P = 1.4/stage) was increased to achieve the required OPR. The other axial-centrifugal configuration had three axial stages (P/P = 1.4/stage) and the pressure ratio of the centrifugal stage was varied to obtain the required OPR. The second axial-centrifugal and the two-stage centrifugal configurations are competitive and result in the minimum mission fuel, acquisition cost, operating cost (500 hrs/yr), and life cycle cost (LCC). For the two-stage centrifugal configuration, the optimum SLS design point pressure ratio based on minimum operating cost is 12.5 for a TIT of 2500° F (2960° R).

Turbine rotor-inlet temperature. - The two-stage centrifugal stage compressor configuration was used to determine the effect of the SLS design point TIT on the various figures of merit for the single-engine light helicopter, figure 7. Compared to the highest TIT considered (2960° R), a TIT of 2300° F (2760° R) results in only about a 0.5 percent increase in each of the figures of merit. Based on 1988 technology this was considered to be a reasonable compromise. This temperature is almost equal to the maximum power TIT for the turboprop (2700° R). Sized at the cruise point, the TS engine requires only about 60 percent of the maximum available power and, therefore, a lower TIT than for the TP. The optimum pressure ratio for the two-stage centrifugal compressor for a TIT of 2760° R is 12.2 based on minimizing the operating cost.

Turbine configuration. - Several combinations of HPT and LPT configurations were studies for the two-stage centrifugal compressor (P/P = 12), figure 8. Both one- and two-stage axial and one-stage radial turbines were considered for the HPT. For the LPT, one- and two-stage axial turbines were investigated. Based on operating cost as the figure of merit, a single-stage radial HPT and a two-stage axial LPT were selected as the optimum turbine configurations for the single-engine light helicopter. Compared with a two-stage axial HPT, a radial HPT results in a 3 percent lower operating cost for this application versus only 1 percent for the light twin fixed-wing aircraft. Therefore, the selected optimum turboshaft engine for the single-engine helicopter has a two-stage centrifugal compressor (P/P = 12 - 60/40 Pr split), a TIT of 2760° R, a single stage radial HPT and a two-stage axial LPT.

Comparison of Various Engines with Advanced Turboprop

Current versus advanced technology TP. - Based on the previously discussed turboprop optimization study, the following engine configuration was selected for determining the effect of advanced engine technology.

Two-spool turboprop
Three-axial plus one-centrifugal stage compressor
Two-stage axial HPT
Two-stage axial LPT

A breakdown of the potential gains in performance which may be realized by means of advanced turbomachinery technology are indicated in Table VII for a 400 HP engine having the above mentioned engine configuration. Relative to a hypothetical current technology turboprop, advanced turbomachinery technology results in a 8.2 percent decrease in ESFC, and a 17.5 percent increase in specific shaft horsepower (shaft horsepower/inlet core airflow) at cruise. The 2 percent increase in compressor efficiency due to advanced technology results in improvements of 1.76 percent in ESFC and 2.87 percent in specific shaft horsepower. The HPT and LPT efficiencies are increased by 1.7 and 2.2 percentage points, respectively. These higher efficiencies are due to improved aerodynamic design, reduced tip clearance (from 2 to 1 percent), and advanced cooling technology. In addition, the advanced cooling technology reduces the turbine cooling requirements for a TIT of 2600° R from 5.8 to 2.8 percent of the compressor exit airflow. As a result, the advanced HPT and LPT reduce the engine ESFC by 3.23 and 2.1 percent while increasing the specific shaft horsepower by 9.63 and 2.66 percent. The remaining projected improvement is obtained by decreasing the overboard leakage from 2 to 1 percent of compressor exit airflow. Thus relative to a current production turboprop, an advanced technology engine could result in a 23 percent lower ESFC.

As indicated in figure 9, the application of advanced engine technology with respect to a light twin engine aircraft results in a significant payoff to all of the figures of merit. The optimum compressor pressure ratio for both a hypothetical engine using current technology and an advanced technology engine is 12 based on minimizing aircraft operating cost. Higher pressure ratios result in reduced mission fuel for the advanced engine. However, the aircraft acquisition cost for both engines increases due to the greater engine cost associated with the higher pressure ratio. Thus both

engines are similar in configuration and cycle. The advanced turboprop with its lower ESFC and lower engine weight (∿20% lower at equal SHP) uses 16 percent less fuel than a hypothetical current technology engine. This combination of improved performance and reduced engine weight results in a 7 percent decrease in aircraft TOGW for the same payload. Being lighter, the aircraft requires less HP. The lower vehicle airframe weight (and, therefore, cost) associated with the lower TOGW plus the lower engine cost (due to the lower HP) for the advanced T
cost by 8 percent. This reduction would be greater if the advanced en cost of $128/HP (based on current production engine cost) could be reduced (as discussed in refs. 2 to 5). The combined effects of improved performance and reduced engine weight for the advanced TP are also reflected in an 8 percent reduction in airplane operating cost based on a utilization rate of 800 hours per year.

Spark ignition recip versus advanced TP. - To penetrate the general aviation light aircraft market, the turboprop must be able to compete with a spark ignition recip (SIR) not only from a performance standpoint, but also from an economic standpoint.

Figure 10 compares a light twin ("rubberized") powered by three difference propulsion systems: (1) an advanced TP, (2) a current production naturally aspirated recip (SIR), and (3) an advanced naturally aspirated recip (SIR). Compared with the current SIR, the advanced SIR has a 10 percent lower BSFC (0.37), a 33 percent lower specific weight (1 lb/HP) and a 2 percentage point higher propeller efficiency; engine specific cost was assumed to be the same (∿30 $/HP - OEM). The advanced turboprop results in a lower TOGW and less mission fuel compared to a production SIR and competitive values of TOGW and mission fuel compared to an advanced technology recip. However, for a turboprop to compete economically, its cost will have to be reduced. To achieve an airplane acquisition cost equal to those for the current and the advanced SIR airplanes, the advanced turboprop OEM cost (nominal of $128/HP based on current production TP costs) will have to be reduced by 50 and 65 percent (specific costs of 64 and 45 $/HP - OEM), respectively. The operating cost for the advanced turboprop is lower than that of the current production SIR engine based on fuel costs of 1.0 and 1.1 $/gal and TBOs of 3000 and 1500 hours for the TP and the SIR engine, respectively. However, to achieve the same operating cost as for the advanced SIR powered airplane, the OEM cost of the TP would have to reduced by 36 percent. References 2, 4, and 5 indicate that cost reductions of about 60 percent are possible for the TP with advanced technology and mass production.

Optimum Engine Summary for All Study Aircraft

Turboprop. - The optimum TP engines for all the twin-engine airplanes were selected on the basis of minimum operating cost; whereas, the optimum engine for the single-engine aircraft was selected on the basis of minimum acquisition cost. The procedure for determining the optimum engine for the light twin (discussed earlier) was repeated for the other missions. The optimum TP engine for each of the fixed-wing aircraft is indicated in table VIII. Except for the horsepower rating, the engine used for the light twin is unchanged for the medium twin. However, the engine for the six-place single was selected based on minimum aircraft acquisition cost which requires sacrificing some performance to minimize engine cost. Therefore, the compressor was changed from an axial-centrifugal to a one-stage centrifugal having a pressure ratio of 9. Also, the HPT was changed from a two-stage axial to a one-stage radial.

Since one purpose of this study is to explore the benefits of advanced technology for turboprops in the light fixed-wing aircraft market, Table VIII indicates the acquisition, operating and LCC for each advanced TP powered, fixed-wing aircraft relative to a current and an advanced SIR engine power aircraft. A number below 1.0 indicates an advantage for the advanced TP. For a number above 1.0, the number in parenthesis indicates the OEM engine cost relative to the calculated current production OEM cost required to achieve cost parity with the SIR engine. As one moves from the medium twin to the high performance single, the advanced TP becomes less competitive with the SIR engines. Therefore, the required reduction in TP cost to achieve cost parity increases. For example, except for acquisition cost, the advanced TP for the medium twin is competitive with both SIR engines. Whereas, for the single-engine aircraft, the OEM cost of the advanced TP would have to be reduced anywhere from 42 to 60 percent depending on the cost parameter, to be competitive with a current SIR engine and from 65 to 74 percent to be competitive with an advanced SIR engine.

Turboshaft. - For the light single-and twin-engine helicopter, the optimum advanced turboshaft engines have the same configuration and cycle. Both engine configurations and cycles were selected on the basis of minimizing operating cost. As indicated in Table VIII, the compressor consists of a two-stage centrifugal compressor (60-40 Pr split) having a pressure ratio of 12. At the sea-level static design point, the TIT is 2760° R. A one-stage radial turbine (HPT) powers the compressor and a two-stage axial turbine (LPT) powers the rotor. Since helicopters, for the most part, are turbine powered, no comparison with a SIR engine was made.

Benefits of advanced technology for each mission. - The benefits of advanced technology as applied to turboprop and turboshaft engines are shown in figure 11 for each of the study aircraft. Indicated are the improvements resulting from advanced turbomachinery and engine weight technology relative to hypothetical current technology turbine engines. Relative to a current production engine the gains would be even greater. Advanced turbomachinery technology refers to higher turbomachinery efficiencies, lower turbine cooling requirements and reduced tip clearances (Appendix C). Advanced turbomachinery technology results in the largest payoffs for each of the figures of merit in figure 11. For example, approximately 90 percent of the mission fuel reduction achieved for each aircraft is due to the advanced turbomachinery technology. The relative importance of each of the turbomachinery technologies is indicated in the sensitivity study (Appendix D).

Of the three fixed-wing aircraft, advanced technology has the largest effect on the six-place light twin. This is due to the differences in the associated improvements in cruise ESFC and engine weight. For the light twin, the improvements in cruise ESFC and engine weight due to advanced technology amount to 10 and 26 percent, respectively. However, for the medium twin these same improvements amount to 8 and 23 percent. The differences in the results for the two helicopters are due to similar effects.

Thus, advanced technology saves upwards of 10 percent in mission fuel at no additional cost. In fact the cost savings could amount to 5 percent.

Perturbations

Effect of takeoff distance. - A majority of the airplanes in the light twin category have takeoff distances over a 35-foot obstacle of between 1575 and 2400 feet (ref. 15). With the advanced turboprop, the light twin has a takeoff distance over a 35-foot obstacle of 1690 feet. This distance is well within the range of takeoff distances of existing airplanes in the light twin category. However, a brief study was made to determine the effect on the figures of merit of decreasing the distance from 1690 (corresponding to the cruise sized engine) to 1608 feet, figure 12. For this study, the wing loading and flap setting (10°) were fixed. As the takeoff distance is decreased below 1690 feet, the engine becomes takeoff-sized and the required horsepower increases by 28 percent. As a result, all of the figures of merit increase as indicated.

Centrifugal compressor efficiency. - A parametric study such as this relies on future trend predictions; consequently, an issue that frequently surfaces is concern over the degree of optimism incorporated into the trend assumptions and the relative importance of these assumptions. The efficiency of future advanced centrifugal compressors is a case in point. Figure 13(a) indicates the nominal curve of efficiency for a centrifugal compressor used in the present study based on discussions with NASA/Lewis compressor specialists. The curve labeled optimistic is based on one of the contracted studies (ref. 3). Parts (b) and (d) of figure 13 indicate the impact of using the optimistic efficiency curve. Cruise BSFC is improved by almost 6 percent as a result of increasing the compressor pressure ratio from 10 (nominal) to 12 (optimistic) due to the increase in compressor efficiency. This also results in a 2.2 percent decrease in operating cost but only a 0.5 percent decrease in aircraft acquisition cost (due to the increased engine cost resulting from the higher OPR).

Centrifugal compressor size correction. - An additional question pertaining to the efficiency of a centrifugal compressor relates to the change in efficiency with corrected airflow or size effect. The insert at the top of figure 14 presents the nominal correction with size used in this study. Also shown is a more optimistic curve used in a contracted study. Applying the optimistic correction to an advanced technology turboshaft having a single-stage centrifugal compressor (point A) powering the light helicopter results in a 3 percent savings in mission fuel. Because of the higher efficiency associated with the optimistic curve, the optimum compressor pressure ratio increases slightly. To better define compressor efficiency, a small component contract study is now underway at Lewis.

Centrifugal compressor cost. - The foregoing results include a generalized cost variation with compressor pressure ratio, but without regard to the type and number of stages (Appendix A). However, it seems reasonable that for conventional designs, compressor cost should vary with part count; thus, a one-stage centrifugal compressor would cost less than a compressor consisting of three-axial plus one-centrifugal stage for a fixed OPR. Exactly how much less the one-stage centrifugal would cost is not known. Based on the calculated OEM engine cost, the compressor accounts for about 9 percent. Figure 15 shows the effect of crediting the one-stage centrifugal compressor with a 50 percent cost reduction relative to a three-axial plus one-centrifugal stage compressor installed in a light twin. Without the part count cost reduction, the choice as to the optimum compressor configuration is relatively clear based on either figure of merit, aircraft acquisition or operating cost. With the 50 percent cost reduction, the one-stage centrifugal configuration is slightly better than the three-axial plus one-centrifugal stage configuration based on aircraft acquisition cost. However, based on operating cost, the figure of merit used for selecting the optimum engine for the fixed-wing light twin aircraft, the three-axial plus one-centrifugal stage compressor configuration continues to be a reasonable choice.

Turbine rotor-inlet temperature factor. - Turbine engine cost (Appendix A) varied with design turbine rotor-inlet temperature ($TIT_d = TIT_{SL} - 100$) according to the line marked nominal in figure 16. The engine cost factor decreases with increasing TIT because the core size required for a given thrust or horsepower is reduced. However, this cost reduction is offset somewhat by the increased cost of the HPT section. The cost adjustment factor is used to reflect both of these changes. The nominal line used in the study decreases by 3 percent as the design TIT is increased from 2100° to 2700° R. The line depicting assumption A decreases by 11 percent over the same temperature range and reflects the cost change for one of the contracted studies. Based on 1988 technology, the design point TIT was limited to 2600° R. For this temperature, assumption A results in aircraft acquisition, operating and LCC cost reductions of 4.5, 1.6, and 1.5 percent, respectively.

CONCLUSIONS

The advanced turbomachinery component technologies forecast for 1988 are estimated to result in the following improvements:

a. A 23 percent decrease in ESFC relative to a current-production turboprop.

b. An 8 percent reduction in ESFC relative to a hypothetical engine using current (1978) technology plus a 20 percent decrease in engine weight.

c. Of this 8 percent decrease in ESFC, approximately 5 percent is attributable to advanced turbine technology.

d. Compared to a hypothetical engine using current technology (1978), the advanced-technology engine results in an airplane fuel savings of approximately 12 percent and cost (acquisition, operating, life cycle) savings of about 5 percent assuming current production engine costs ($/HP).

Compared with an improved spark ignition reciprocating (SIR) engine (10 percent lower BSFC, 30% lower engine specific weight) the advanced turboprop is competitive in terms of fuel savings, but requires engine cost reductions ($/HP - OEM) of 5 to 65 percent (depending on the mission) to achieve operating and life cycle cost parity with the advanced SIR engine.

The advanced turboprop is more cost competitive in larger aircraft (medium twin) and less competitive in lighter aircraft (light twin, single engine).

Contractor studies indicate cost reductions of the magnitude required for turboprops to be competitive with SIR engines can be achieved by means of advanced technology and mass production. This would then permit the many advantages of turboprops (less vibration, safety, higher TBO's, multi-fuel capability, etc.) to be enjoyed by many general aviation owners.

Compared with a hypothetical current technology turboshaft, an advanced technology engine results in a 16 percent savings in fuel and an 11.4 percent savings in life cycle cost when powering a light twin helicopter.

APPENDIX A

ENGINE COSTS

Basic Equations

Turboprop and turbofans. - The turbine engine cost model is a collection of single parameter curve fits based on information contained in references 16-24 which are listed in descending order of importance in the development of these relationships. The baseline year for these cost curves is 1976. Adjustments of data from other years were computed using a seven percent inflation rate. Extrapolation to 1977 dollars, the cost baseline used in this report, was also made by assuming the same 7% annual cost increase.

Although the basic cost relationships used in this cost model are valid for very large engines, the complete cost model including all of the correction factors is valid only for engines under 1000 horsepower or under 1500 lbs. of thrust.

The basic OEM price curves for turbofans and turboprops are shown in figures A1 and A2 respectively. The curves were checked against a range of engine sizes varying from engines for large, wide-body aircraft to engines for small missles. All of the engines checked fell within the original band of data used to develop these curves. These curves represent the typical engine being built as of 1976: similar in technology, unit weight, specific fuel consumption, etc.. To compare these relationships with the price of any one actual engine might be misleading. A given real engine might be far from typical, and for a fair comparison the differences must be known, even the production rate. Correction relationships are presented in the last section of this appendix. Most of these correction factors are only valid for small engines.

Spark ignition reciprocating (SIR). - The SIR engine cost curve presented in figure A2 is only valid below 1000 horsepower. This model assumes that all engines above 300 horsepower are turbocharged. Ordinarily for a given basic engine configuration the addition of turbocharging would add to the price. However, the spread in engine price for engines of the same horsepower and configuration is greater than the increase in going from non-turbocharged to turbocharged. Therefore no step increase in cost is assumed at the 300 horsepower transition point where this change is assumed to take place.

The curve of propeller cost as a function of shaft horsepower, presented in figure A3, represents current-production technology. It is the same model used in the original GASP computer program (ref. 6) except that the curve fit is modified at low propeller weights. This modification was necessary since the original GASP relationship predicted negative prices at very small weights. The model used in this study assumes increasing complexity and, therefore, a higher cost per pound as weight increases.

Correction Multipliers

Correction curves are used to modify the basic engine cost of turbofan and turboprop engines which were obtained from the relationships presented in the preceding section. The corrections account for changes in turbine engine cost with changes in bypass ratio, overall pressure ratio and turbine rotor-inlet temperature. However, except for the HPT, engine cost is not affected by part count or by the type of compressor (axial, axial-centrifugal, centrifugal) or the type of turbine (radial, axial).

Bypass ratio. - Figure A-4 presents the correction for bypass ratio for a turbofan. This curve is used to account for the fact that as the bypass ratio increases with a resulting increase in fan and engine diameter the cost of the engine increases, although the core size decreases.

Overall pressure ratio. - The correction factors for overall pressure ratio are presented in figure A-5. There are two curves here. The upper represents the factors for turbine engines with multiple stage high pressure turbines. The lower curve is for single-stage high pressure turbine configurations. Engines with a multi-stage high pressure turbine are about 5% higher in cost than those with a single stage HPT.

Turbine temperature. - The correction factor for turbine rotor inlet temperature, TIT, is presented in figure A-6. The curve appears to be anomolous since it predicts a decrease in engine cost as the turbine inlet temperature increases. Two factors offset the increase in hot section cost and cause the negative slope of the curve. The first is fundamental. The airflow required for a given thrust or output power, and thus the size and weight of the engine, decrease when TIT is increased. The second factor is a result of the method of the cost model development. The turbine inlet temperature correction curve is only meaningful when used together with figure A-5. For optimum engines if a higher TIT is specified, a higher compression ratio is usually required. In figure A-5 the stress and parts-count increase predominate, while in figure A-6 the size reduction predominates.

The temperature correction curve is only valid up to 3000° R (2540° F). Above that temperature, complex cooling methods or the use of very advanced materials would have be included in the model.

APPENDIX B

HELICOPTER COSTS

The helicopter cost model was developed from several sources. The major helicopter cost relationships are presented in this appendix except for engine cost relationships which are presented in Appendix A.

The relationships for airframe costs were obtained by combining a weight equation with cost equations from the Zodiac II helicopter design code, reference 25. All costs have been adjusted to 1977 dollars assuming a constant 7 percent rate of inflation.

The airframe cost is made up of three parts: labor cost, materials cost, and engineering and testing costs.

The relationship for these three parts are:

$$\text{LABOR COST} = 1418[0.749\ We + 126]^{0.85}\ [TNP^{-0.39}] \tag{B-1}$$

$$\text{MATERIALS COST} = [9.7819\text{x}10^{-2}\ We + 1.6456][TAS^{1.24}][TNP^{-0.12}] \tag{B-2}$$

$$\text{ENGINEERING \& TEST COSTS} = [We + 168.2243][279.6317\ (TNP^{-1}) + 0.9533\ (TNP^{-0.15})] \tag{B-3}$$

Where

We is empty weight in pounds

TNP is total number produced and

TAS is the maximum true air speed in miles per hour.

The transmission price relationship was developed from information in reference 26. In 1977 dollars this equation is,

$$\text{TRANSMISSION PRICE} = 4.44364\ (\text{GROSS WEIGHT})^{1.04263} \tag{B-4}$$

where gross weight is in pounds.

Reference 27 suggests that there is a great scatter in avionics costs, rotor costs, and miscellaneous costs. Avionics cost is a user input. However rotor and miscellaneous costs are estimated from information in reference 27. The relationships are as follows:

$$\text{ROTOR COST} = 0.1294\ (\text{TOTAL AIRCRAFT COST}) \tag{B-5}$$

$$\text{MISC. COSTS} = 0.2353\ (\text{TOTAL AIRCRAFT COST}) \tag{B-6}$$

Equations (B5 and B6) imply that the airframe, engines, and transmissions make up 63.53% of the total aircraft cost and the rotor and miscellaneous items make up the remaining 36.47%.

From reference 27 a maintenance cost relationship was developed for the non-propulsion sections of the aircraft. In 1977 dollars the equation is,

$$\text{NONPROPULSIVE MAINTENANCE} = 0.77423[3.4205(\text{GROSS WEIGHT})^{1.56977} + 135.81] \tag{B-7}$$

APPENDIX C

TURBOMACHINERY TECHNOLOGY

Two levels of turbomachinery technology pertaining to efficiency and cooling were assumed. These levels represent current (1978) and advanced (1988) technology. Both are based on available data (most related to current technology) and discussions with NASA component specialists. First, the baseline efficiency for each component is discussed. Next the efficiency adjustments for tip clearance, size, and cooling effects are discussed.

Fan Efficiency

The basic adiabatic single-stage fan efficiencies (ηT-T) are shown in figure 1-C(a). These efficiencies are based on a fan having an inlet corrected-airflow of ‘0 lb/sec.

Axial Compressor Efficiency

Single stage axial compressor polytropic efficiency (ηp) is presented in figure 1-C(b) as a function of pressure ratio for a corrected airflow of 10 lb/sec. An axial compressor can consist of any number of axial stages. All stages are assumed to have the same pressure ratio and polytropic efficiency. The basis adiabatic efficiency of each axial stage is calcuated according to the following equation:

$$\eta_{ad} = \frac{(P/P)^{(\gamma-1)/\gamma} - 1}{(P/P)^{(\gamma-1)/(\gamma\eta_p)} - 1}$$

East stage efficiency is then corrected for size effect. The overall compressor adiabatic efficiency is then calculated as follows:

$$\eta_c = \frac{(P/P)^{(\gamma-1)/\gamma} - 1}{\prod_{j=1}^{N}\left[1 + \frac{1}{\eta_{cj}}\left\{(P/P)_j^{(\gamma-1)/\gamma} - 1\right\}\right] - 1}$$

Centrifugal Compressor Efficiency

Adiabatic efficiencies used for one and two stage centrifugal compressors are shown in figure 1-C(c) for a compressor inlet corrected airflow of 8 lb/sec. For a two-stage centrifugal compressor a 60/40 pressure ratio split is used and a 2 percent interstage total pressure loss is assumed. If both centrifugal stages are on the same spool, each of the stages is also penalized 1/2 percent to account for their non-optimum speeds.

Axial Turbine Efficiencies

The stage adiabatic efficiencies for both high pressure (HPT) and low pressure (LPT) turbine are indicated in figure 2-C(a). These efficiencies are plotted as a function of stage work factor ($gJ\Delta H/stage/Um^2$) (Um-mean blade velocity) for a corrected inlet gas flow of 5 lb/sec. For the HPT and LPT, the mean blade velocities are assumed to be 1350 and 1200 ft/sec. (based on an average of existing engines). These efficiencies are for uncooled turbines have zero tip clearance.

Radial Turbines

The efficiency of a radial inflow-turbine is shown in figure 2-C(b) as a function of stage work factor ($gJ\Delta H$ stage/Ut^2 were Ut is the tip velocity). Tip velocities of 1800 and 1900 ft/sec are used for current and advanced turbines, respectively. The efficiencies are based on typical tip clearances and an uncooled stage.

Turbine Tip Clearance Correction

Tip clearances of 2 and 1 percent (of blade height) are used for the current and advanced axial turbines, respectively. The associated decrements in axial turbine efficiency are indicated in figure 3-C(a). These corrections are based on a flash wall.

Component Size Correction

The effect of size on component efficiency is presented in figure 3-C(b) as a function of stage inlet corrected-flow. These corrections are subtracted from the nominal component efficiencies of figure 1-C and 2-C. The corrections are based on discussions with NASA component personnel.

Turbine Cooling

Because of the requirement to cool turbine blades, cooling passages must be incorporated. The associated increase in blade thickness and the exiting of the coolant from the blade reduces the turbine efficiency.

Turbine cooling requirements assumed in this study are shown in figure 4-C(a). The upper curve, based on full-coverage film-cooling, is used for the advanced engines. The lower curve, based on blade insert cooling with the coolant exiting at the blade trailing edge is used for the current technology engines. The terms in the ordinate are defined as follows:

Tg	turbine gas temperature, ^{o}R
Tm	allowable metal temperature, ^{o}R
Tc	coolant temperature, ^{o}R

The decrease in turbine stage efficiency due to cooling is accountered for by means of a correction factor $(\Delta\eta)/(\eta Wc/Wg)$. As shown in figure 4-C(b) the value of this factor varies with vane, blade, and year of technology.

Turbine cooling is discussed in more detail in the ANALYSIS portion of this report.

APPENDIX D

SENSITIVITY STUDY

The sensitivity studies were included as part of the analytical studies in order to determine which engine parameters have the greatest impact on the important aircraft figures of merit. In this study four figures of merit were considered, mission fuel, aircraft acquisition cost, operating cost and life cycle cost. Those parameters with the greatest impact on the aircraft figures of merit represent both possible pitfalls in engine design and possible areas of opportunity for engine research.

One fixed wing aircraft, the light twin powered with optimum advanced turboprop engines, and one helicopter, the single-engine light aircraft powered with an optimum advanced turboshaft engine were studied in detail. A few points were run for other configurations. It was found tnat although the absolute levels of the aircraft figures of merit change greatly from one aircraft to another, the percent change in a given figure of merit for a given percent change in an engine parameter varies very little from aircraft to aircraft.

Light Twin Airplane

Percent changes in mission fuel, acquisition cost, operating cost and life cycle cost due to variations in compressor pressure ratio, compressor efficiency, high pressure turbine (HPT) and low pressure turbine (LPT) efficiencies, and engine bleed (leakage and/or pressurization) are shown for the light twin airplane in figures D1 through D5.

These figures show that fuel consumption, operating cost and life cycle cost are slightly more sensitive to changes in efficiency for the HPT than for the compressor or the LPT. Acquisition cost is more sensitive to compressor efficiency. Compressor pressure ratio and engine bleed have the least impact on a percent increase basis. It is possible, however, that required engine bleed could be several times that of the 1% of core flow of the base case dependent upon changes in altitude capability and pressurization requirements. Figure D5 presents the effect of a 300% engine bleed increase from a 1% to a 4% bleed.

Helicopter

Figures D6-D10 present similar information for the light, single-engine helicopter. A sensitivity analysis similar to that for the light, fixed-wing twin was done for the light, single-engine helicopter. In addition to engine parameters examined in the fixed-wing analysis, the effects of changes in cooling bleed and power extraction were also studied.

The figures show that the important parameters are still compressor efficiency and high-pressure and low-pressure turbine efficiencies. Unlike the results for the fixed wing aircraft, the light helicopter results indicate that the compressor efficiency is always the parameter having the greatest impact on the key variables. Changes in engine bleed, cooling bleed, and compressor pressure ratio turned out too small to plot. Note, too, that operating cost and life cycle cost sensitivities turned out to be exactly the same for the helicopter.

General

Although not exactly the same, the closeness of the sensitivity results for the helicopter and those of the light twin for any given parameter indicate why sensitivity curves for all aircraft were not included in this study.

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TABLE I

REPRESENTATIVE CURRENT SMALL ENGINE

TURBOPROP

ENGINE	YEAR CERTIFIED	MAX. S. L. THERMO ESHP	SLS AIRFLOW	MAX TIT	COMPRESSOR P/P	CONFIG. * (COMP.)(T)(T)	DRY WEIGHT	SLS ESFC
PT6A-11	1977	528	6.8 lb/sec	2380 °R	6	(3A/C) (A)(A)	300 lbs.	.65 lb/hp/hr
TPE-331-1	1967	705	6.2	2300	8.34	(C/C)(3A)	336	.60
250B-17B	1974	417	3.4	2457	7.2	(6A/C) (2A)(2A)	195	.63
LTP 101-600	1976	620	4.8	2345	8.5	(A/C) (1A) (1A)	325	.55

TURBOSHAFT

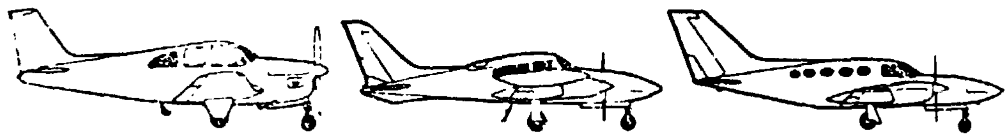
ENGINE	YEAR CERTIFIED	MAX. S. L. THERMO ESHP	SLS AIRFLOW	MAX TIT	COMPRESSOR P/P	CONFIG. * (COMP.)(T)(T)	DRY WEIGHT	SLS ESFC
LTS 101-600	1973	615	4.8	2345	8.5	(A/C) (1A)(1A)	241	.57
250 C-30	1978	700	5.6	2514	8.5	(C) (2A)(2A)	235	.59

* A/C denotes axial-centrifugal

TABLE II

STUDY AIRCRAFT

FIXED-WING



PARAMETER \ CATEGORY	6-PLACE Hi-perf. Single Engine	6-PLACE Light Twin-Engine	6-PLACE Medium Twin-Engine
Similar Aircraft	Bonanza	310	421
Payload, lb	1000	1000	1200
Cruise Altitude, ft	7500	10000	25000
Cruise Speed, Kts	174	226	234
Range, N. Mi (100% load factor)	500	1100	1370
Wing Loading, lb/ft^2	25	29	35
Aspect Ratio	8	8	7.37
Sweep @ 1/4C	0	0	0
Flaps	Single Slotted	Single Slotted	Single Slotted
Number Engines	1	2	2
Takeoff Field Length, ft (over 35 ft.)	< 1800	< 2000	< 2200
Pressurized	No	No	Yes (8000 ft.)

PARAMETER \ CATEGORY	4-PLACE Light Single Helicopter	6-PLACE Light Twin Helicopter
Payload, lb	800	1200
Cruise Altitude, ft	SL	SL
Cruise Speed, Kts	110	125
Range, N. Mi.	300	450
Disc Loading, SHP/A	4	5.5
Hover Ceiling, ft (Out of Ground Effect)	6000	8000

TABLE III

CRUISE COMPARISION OF SMALL TURBOPROP ENGINES

(20000 ft and 217 Kts)

	Current Production Allison 250-B17B (420 SHP @ SLS)	Hypothetical Engine Using Currently Available Technology
Pressure ratio	7.2	12
Turbine rotor-inlet temp. ^{O}R	2457	2600
SHP, HP	225	225
ESHP, HP	244	244
BSFC, lb/HP/hr	.58	.495
ESFC, lb/HP/hr	.56	.476
Uninstalled Weight, lb.*	224	208
Specific weight, lb/HP*	.56	.52

*Predicted by GASP program. Actual 250-B17B dry weight is 195 lb.

TABLE IV

RECIP AND TURBOPROP COMPARISON

	Reciprocating Turbocharged	Hypothetical Turboprop Using Currently Available Technology
SHP_{SLS}, HP	428	325
Weight, lb/HP	1.66	.625
Cruise at 25000 ft/234 Kts.		
--eshp, HP	257	185
--esfc, lb/HP/hr	.45	.462
Installation Losses		
--Cooling drag, % C_D	10	0
--Nacelle drag, % C_{DO}	5	3
--Propeller efficiency	.87	.89
Fuel - Type	Avgas	MULTIFUEL
$/gal.	1.1	1.0
TBO, hrs	1200	3000
Cost $/eshp (OEM)	32.4	117.8
Engine Operating Cost		
Maint. $/flt-hr	6.3	7.1
Overhaul, $/flt-hr	7.2	7.9

TABLE V

MISSION COMPARISON BASED ON TWO DIFFERENT POWERPLANTS

SIX PASSENGER MEDIUM TWIN

CRUISE AT 234 KTS AND 25000 FT FOR 1,370 N. Mi

Engine Size, SHP, HP	Turbocharged Reciprocating	Hypothetical Turboprop Using Currently Available Technology
--Cruise	257	185
--Takeoff	428	325
Weights, lb		
--Engines	1417	410
--Fuel used	1438	1063
--Engine + fuel	2855	1473
--A/C Empty wt	4688	3274
--TOGW	7841	6002
Cost, $		
--Engines (OEM), both	27698	76032
--Airframe (OEM)	53265	45555
--Total	203116	290899
--Operating, $/ft hr. (800 hr/yr)	114	109
--5 Year life cycle	462573	445409

TABLE VI

ADVANCED TURBOPROP vs ADVANCED TURBOFAN
25000 FT/234 Kts

Parameter	Turboprop	Turbofan
Fan pressure ratio	---	1.5
Compressor pressure ratio	12	12
Turbine rotor inlet temperature, ^{O}R	2600	2600
Bypass ratio	---	8
Thrust, lb	318	318
TSFC, $\frac{lb}{lb\text{-}hr}$	.331	.518
Wa, CORE Lb/sec	1.12	2.07

TABLE VII

SMALL TURBINE ENGINE COMPONENT IMPROVEMENT

400 HP SLS SIZE

DESIGN POINT 234 KTS - 25000 FT

	1978 Baseline Turboprop	1988 Forecast Δ's Change	% ESFC	% SHP/Wa
Compressor				
η	.789	+ .02[illegible]	- 1.76	+ 2.87
(P/P)	12.	0		
Combustor				
η	.985	0	0	C
ΔP/P	.04	0	0	0
Gas Generator Turbine (HPT)				
η	.838	+ .017	- 3.23	+ 9.63
P/P	3.41	- .38		
TIT, ^{o}R	2600	0		
Cooling Bleed, %	5.8	-3.0		
Power Turbine (LPT)				
η	.877	+ .022	- 2.1	+ 2.66
P/P	3.41	+ .43		
Shaft & Bearings η	.99	0	0	0
Gearbox η	.99	0	0	0
Overboard Leakage, %	2	-1	- 1.1	+ 2.30
TOTAL ENGINE			- 8.2	+ 17.5

TABLE VIII

SUMMARY OF VARIOUS AIRCRAFT USING ADVANCED TURBINE ENGINES

CONFIGURATION	6 PLACE MEDIUM TWIN	6 PLACE LIGHT TWIN	6 PLACE HIGH PERF SINGLE	LIGHT SINGLE HELICOPTER	LIGHT TWIN HELICOPTER
AIRCRAFT					
Gross weight, lbs	5636	5150	2957	2119	4275
Payload, lb	1200	1000	1000	800	1200
cruise vel., kts	234	226	174	110	125
Range, N. Mi.	1370	1100	500	300	450
OPT. ENGINE DESIGN					
compressor configuration	3A + 1C	→	1-C	2C	→
compressor pressure ratio	12	→	9	12	→
TIT, °R	2600		→	2760	→
HPT configuration	2A	→	1R		→
LPT configuration	2A				→
SLS SHP, HP	307	226	186	273	327
cruise ESFC, lb./HP./hr	.426	.447	.493	.476	.464
COST RATIO TP/CURRENT RECIP*					
Acquisition	1.37 (.48)	1.36 (.50)	1.47 (.40)		
Operating	.91 (>1.0)	.95 (>1.0)	1.18 (.58)		
LCC	.91 (>1.0)	.95 (>1.0)	1.18 (.58)		
COST RATIO TP/ADVANCED RECIP*					
Acquisition	1.49 (.33)	1.53 (.35)	1.65 (.26)		
Operating	1.02 (.94)	1.10 (.64)	1.30 (.36)		
LCC	1.03 (.91)	1.10 (.70)	1.30 (.35)		

*The values listed are the costs of turbine-powered aircraft divided by the cost of recip aircraft.

The numbers in ( ) are the relative turbine engine costs to achieve cost parity with recip aircraft.

A Axial
C Centrifugal
R Radial

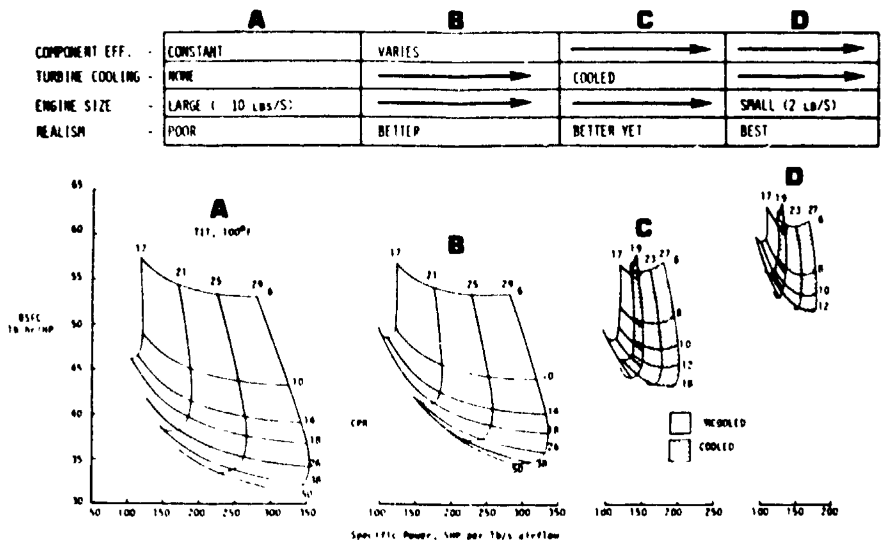


Figure 1 Cycle parametric effects on small engine performance

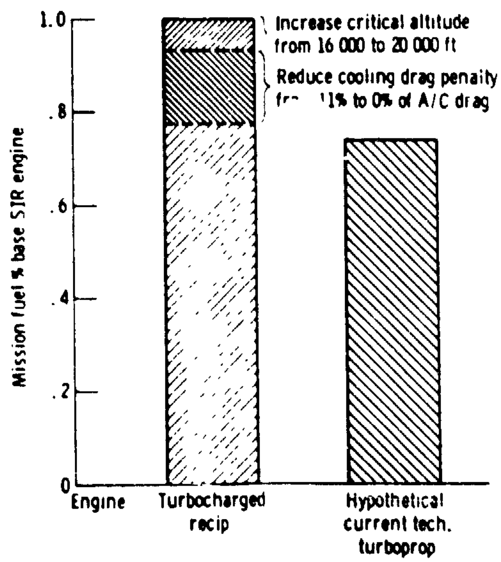
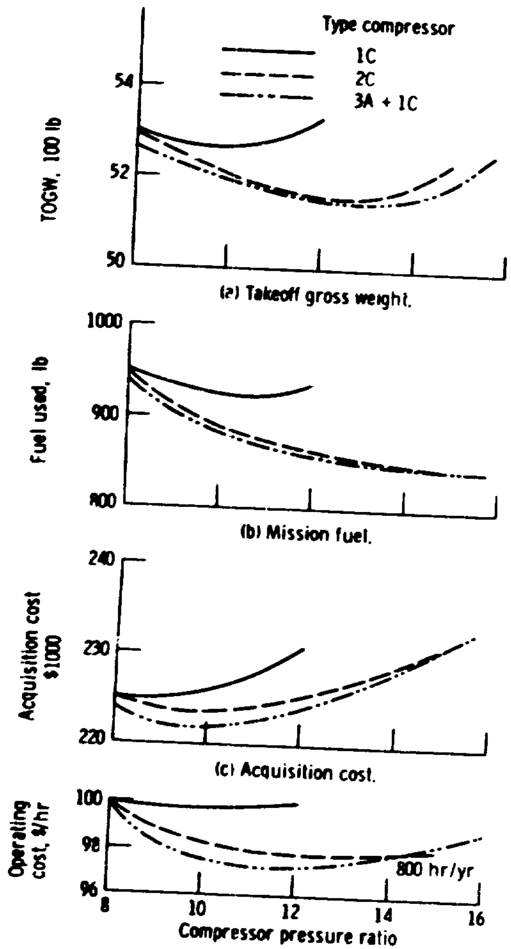


Figure 2 - Effect on mission fuel of increasing the critical altitude of a turbocharged spark ignition reciprocating engine and reducing the engine cooling drag Medium twin-engine airplane. Range - 1370 n. mi. 234 knots at 25 000 ft.



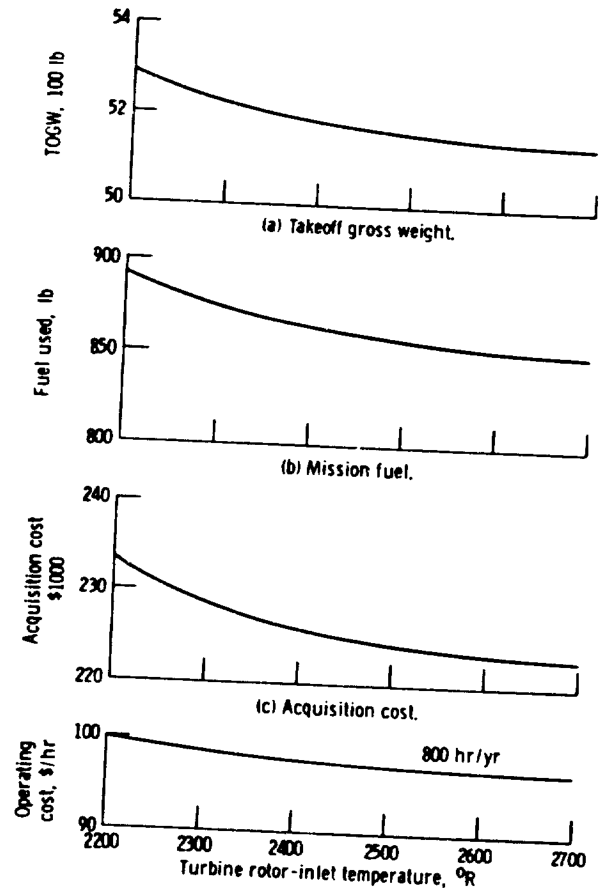
(a) Takeoff gross weight.

(b) Mission fuel.

(c) Acquisition cost.

(d) Operating cost.

Figure 3. - Effect of pressure ratio and type of compressor on a 6-place advanced twin-engine TP airplane. R = 1100 n. mi.; 226 knots at 10 000 ft; twin-spool free turbines; TIT = 2600° R; 2A-HPT; 2A-LPT; 60/40 pressure ratio split on the C-C; 3 axial Pr = 2.2.



(a) Takeoff gross weight.

(b) Mission fuel.

(c) Acquisition cost.

(d) Operating cost.

Figure 4. - Effect of HPT turbine rotor-inlet temperature on a 6-place advanced twin-engine TP airplane. R = 1100 n. mi.; 226 knots at 10 000 ft; twin-spool free turbine; 2A-HPT; 2A-LPT; 3 axial + 1-centrifugal stage compressor. Pr_{axial} = 2.2; OPR = 12.

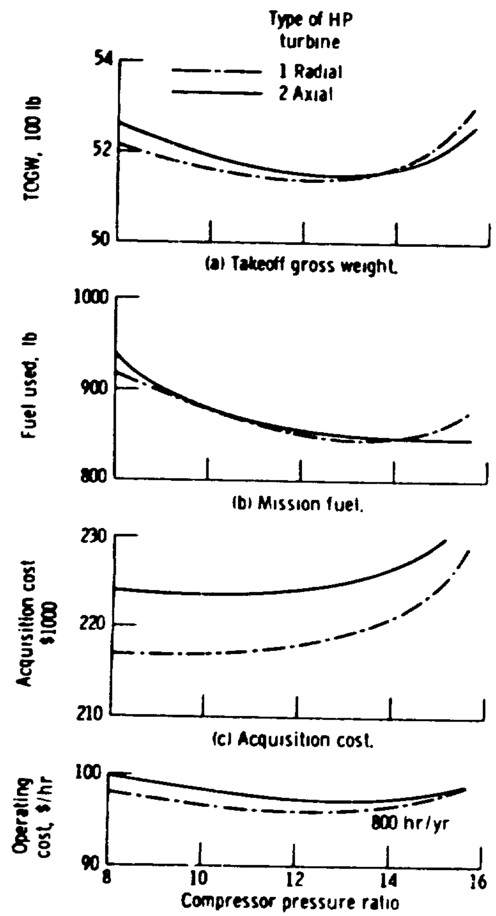


Figure 5. - Effect of the type of high pressure turbine on a 6-place advanced twin-engine TP airplane. R = 1100 n. mi.; 226 knots at 1000 ft; twin-spool free turbine; TIT = 2600° R; 2A-LPT; 3-stage axial + 1 stage centrifugal compressor; Pr_A = 2.2.

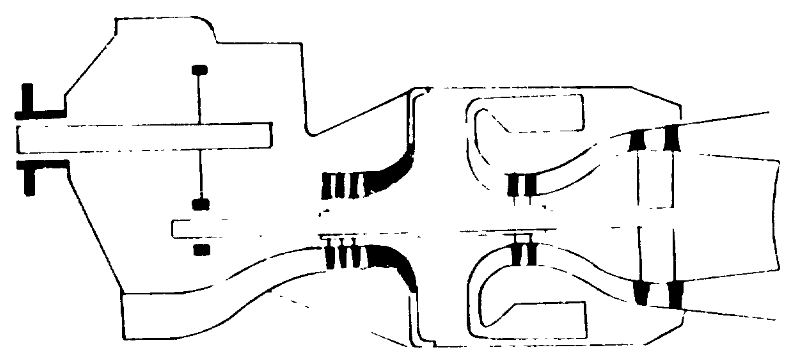


Figure 6. - Advanced engine configuration.

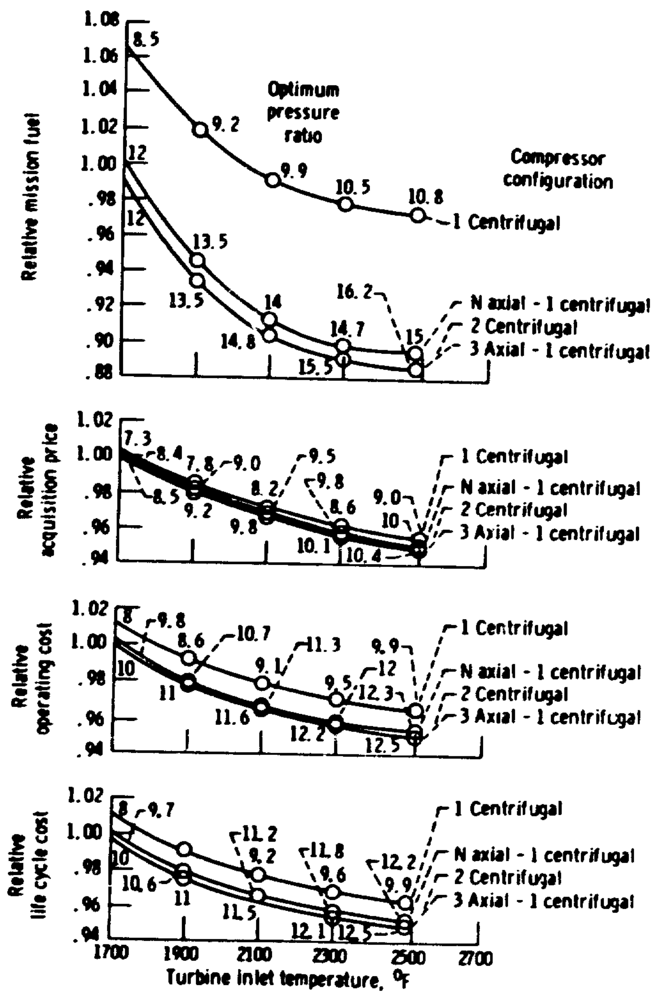


Figure 7. - Effect of compressor configuration and turbine inlet temperature on various figures of merit.

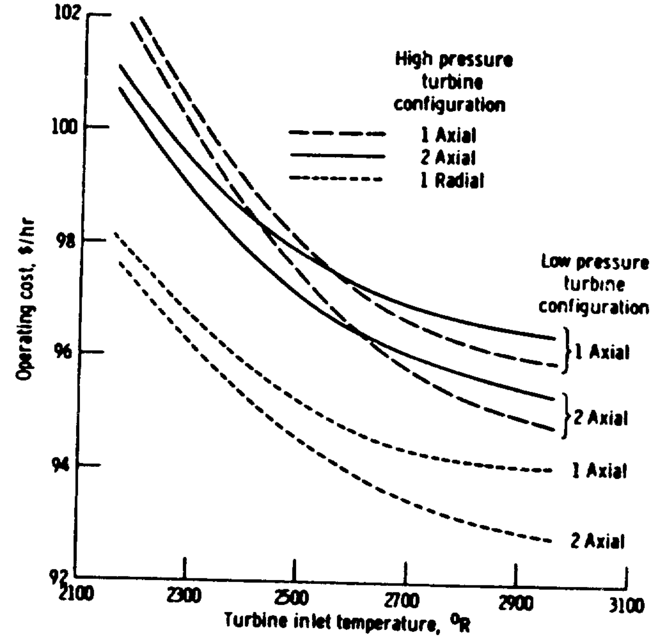


Figure 8. - Effect of the type of high and low pressure turbine on a single-engine light helicopter. Two-stage centrifugal compressor, P/P - 12

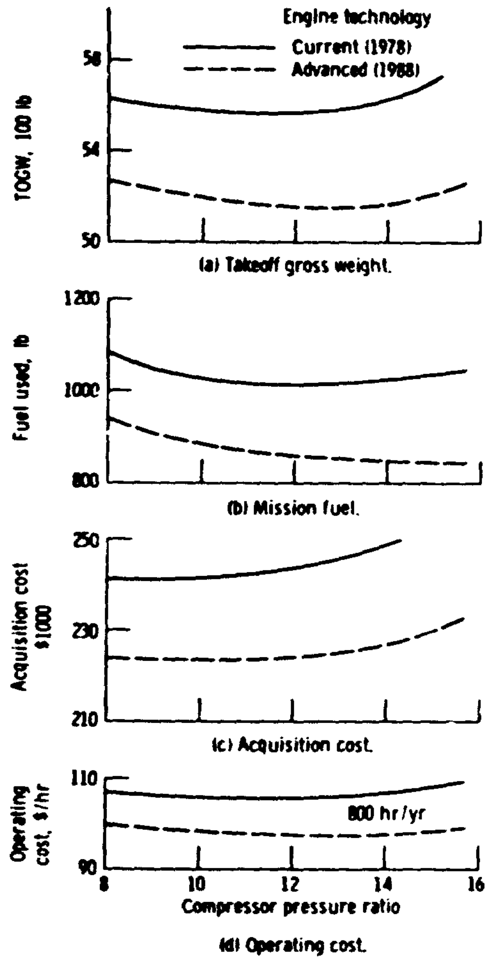


Figure 9. - Effect of engine technology on a 6-place twin-engine TP airplane. Range - 1100 n. mi.; 226 knots at 10 000 ft; twin-spool free turbine; TIT - 2600° R; 2A-HPT; 2A-LPT; 3 axial stages (Pr - 2.2) - 1 stage centrifugal compressor.

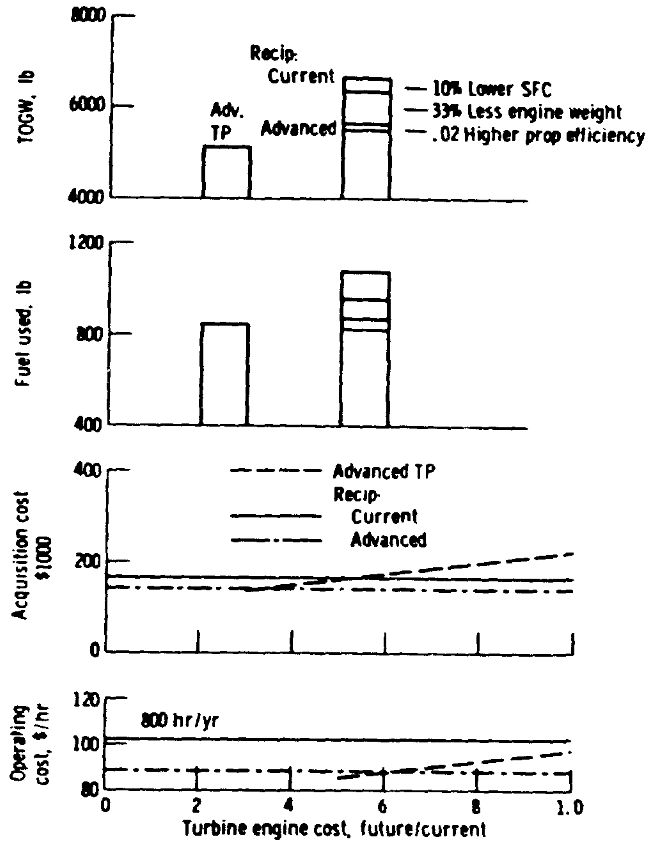


Figure 10. - Advanced TP versus a reciprocating power-plant for a light twin engine airplane. R - 1100 n. mi.; 226 knots at 10 000 ft; twin-spool free turbine; TIT - 2600° R; 2A-HPT; 2A-LPT; 3-axial plus 1-centrifugal compressor. Naturally aspirated recip.

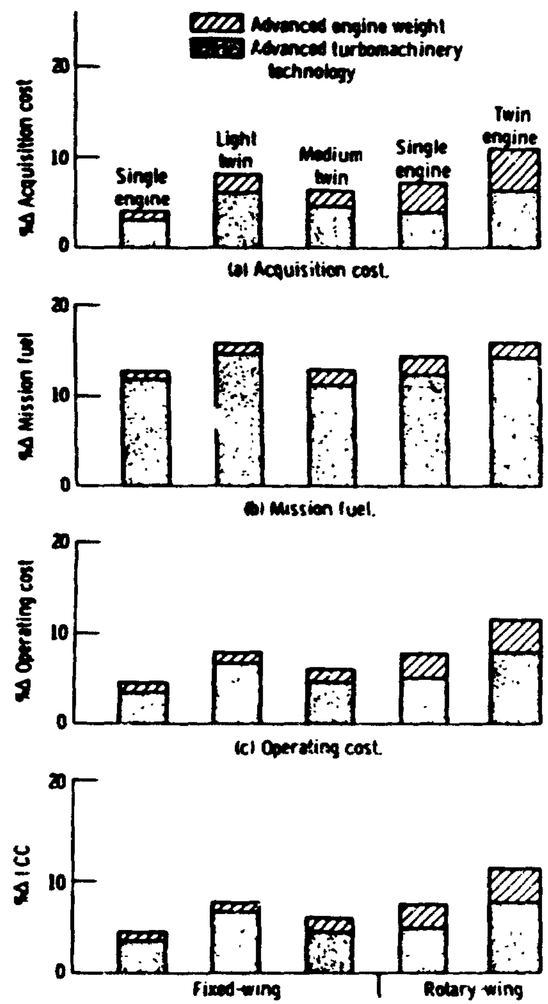


Figure 11. - Effect of advanced technology.

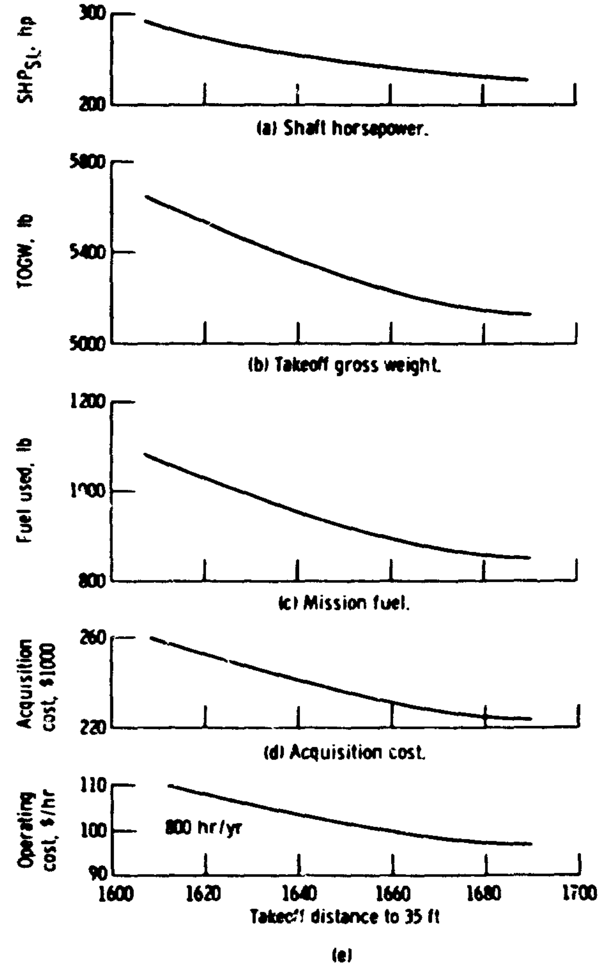


Figure 12. - Effect of takeoff distance on a 6-place advanced light twin-engine TP airplane. R - 1100 n. mi.; 226 knots at 10 000 ft; twin-spool free turbine; TIT - 2600° R; 2A-HPT; 2A-LPT; 3 stage axial + 1 stage centrifugal compressor; OPR - 12.

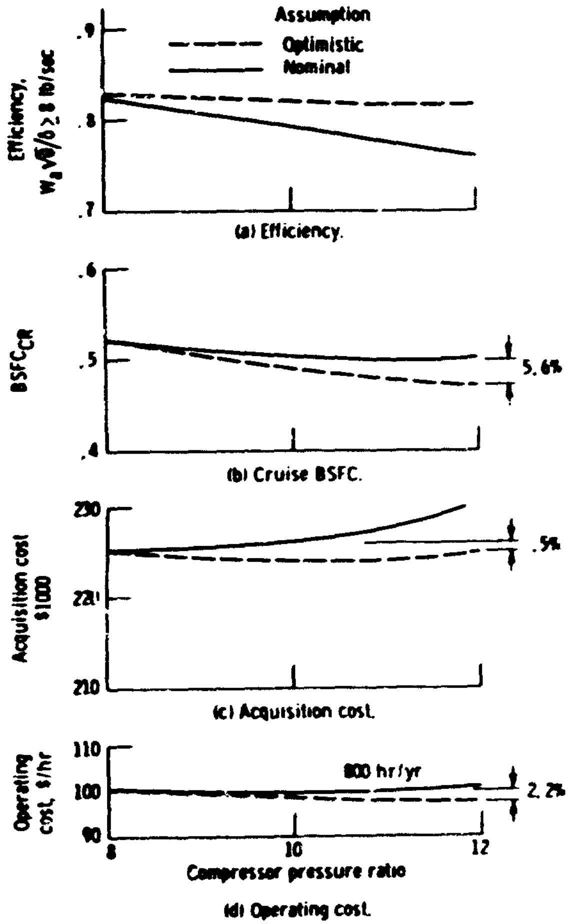


Figure 13. - Effect of centrifugal compressor efficiency assumption on a light twin-engine TP, 6-place airplane. R - 1100 n. mi.; 226 knots at 10 000 ft; twin-spool free turbine; 2A-HPT; 2A-LPT; TIT - 2600° R.

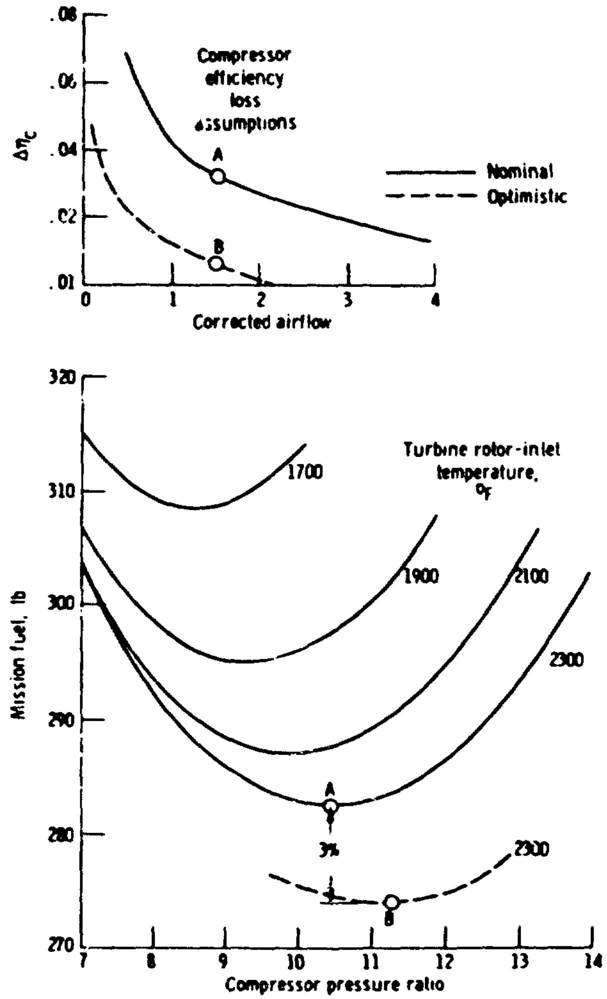


Figure 14. - Effect of centrifugal compressor efficiency airflow (size) correction factor on mission fuel for a single-engine light helicopter

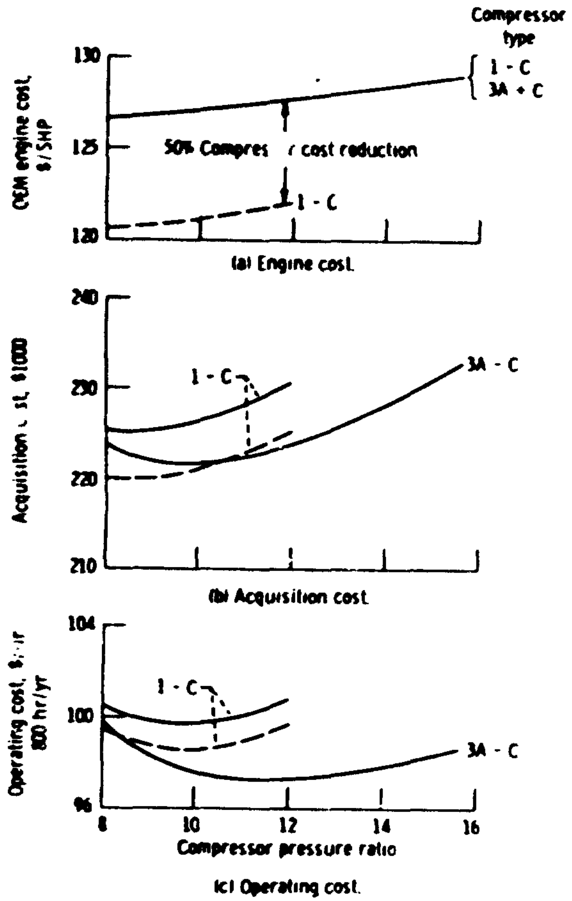


Figure 15. - Effect of reduced compressor cost on a 6-place light twin-engine airplane. R = 1100 n. mi., 226 knots at 10 000 ft; twin-spool free turbine, TIT = 2600° R; 2A-HPT, 2A-LPT, Pr_{3A} = 2.2.

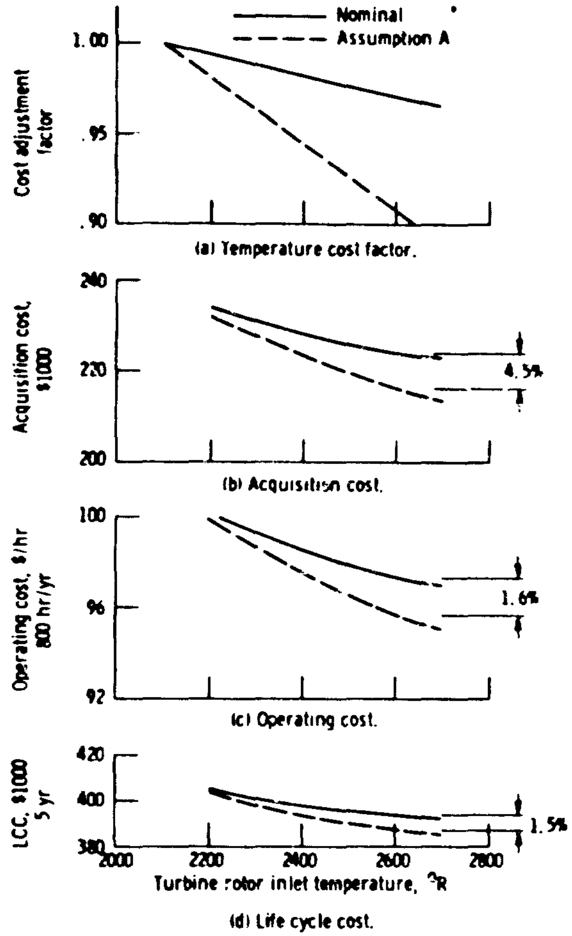


Figure 16. - Effect of turbine rotor-inlet temperature cost factor. Advanced light twin-engine TP airplane; R = 1100 n. mi.; 226 knots at 10 000 ft, twin-spool free turbine; TIT = 2600° R, 2A-HPT, 2A-LPT; 3 axial + 1-centrifugal compressor; Pr_{3A} = 2.2.

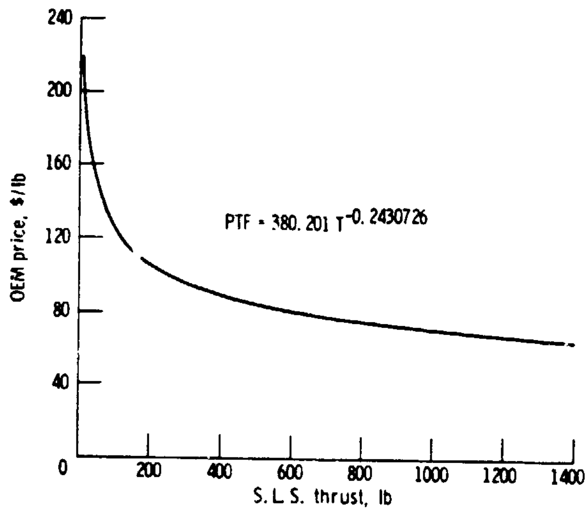


Figure A1. - Basic OEM unit price for turbofans (1976 dollars).

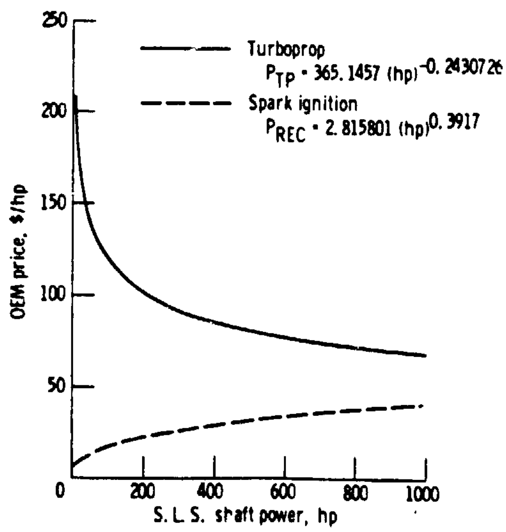


Figure A2. - Basic OEM unit price for turboprops and reciprocating engines (1976 dollars).

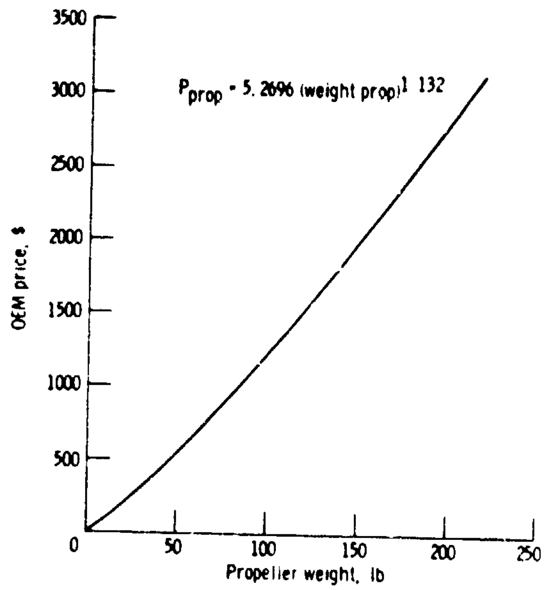


Figure A3. - Basic OEM price for propellers (1976 dollars).

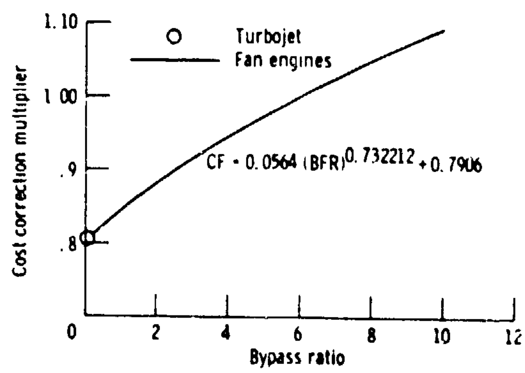


Figure A4. - Bypass ratio cost correction factor.

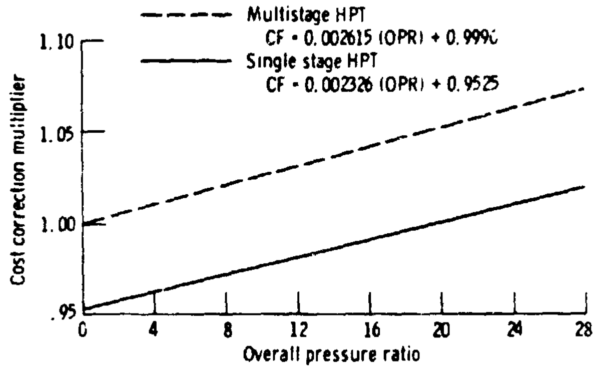


Figure A5. - Overall pressure ratio, high pressure turbine configuration cost correction factor

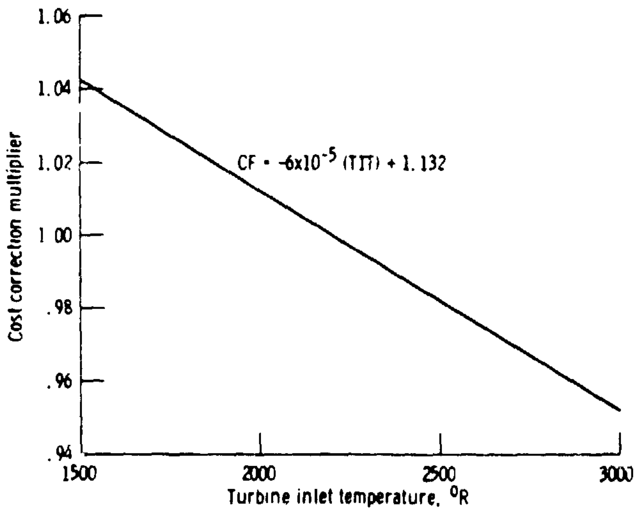
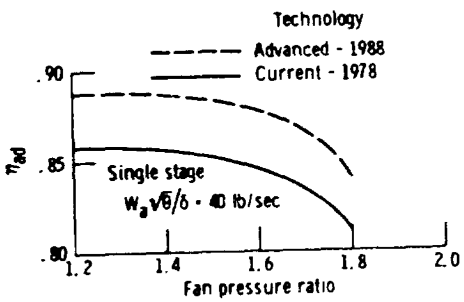
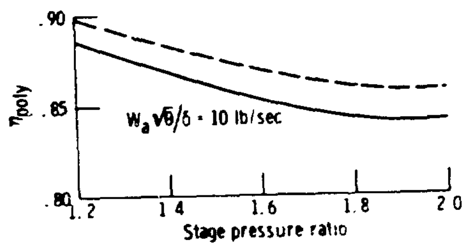


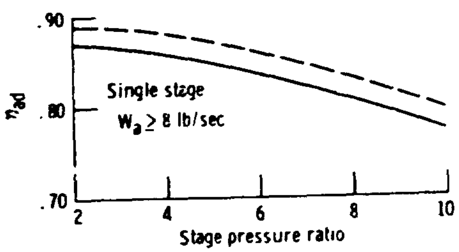
Figure A6. - Cost correction factor for turbine inlet temperature (TIT).



(a) Fan adiabatic efficiency.

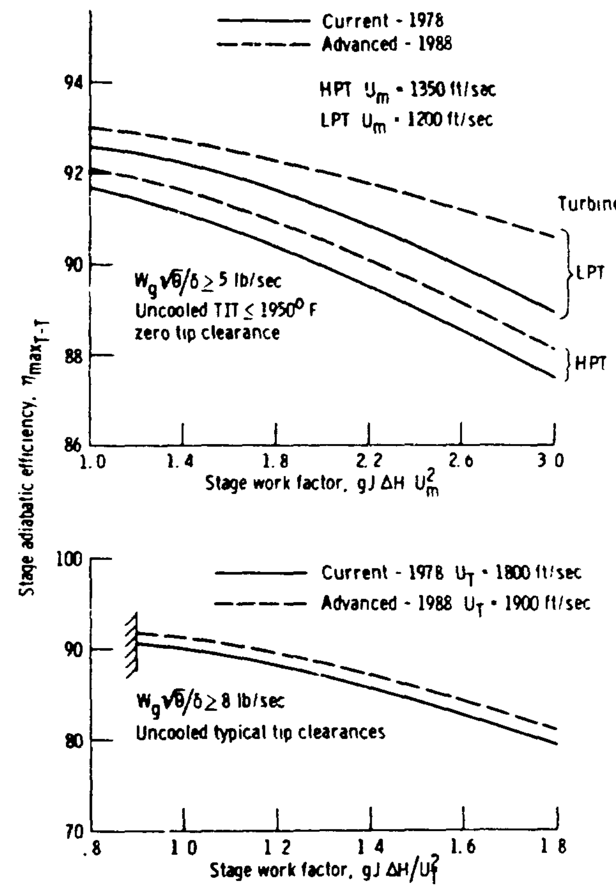


(b) Axial compressor stage polytropic efficiency



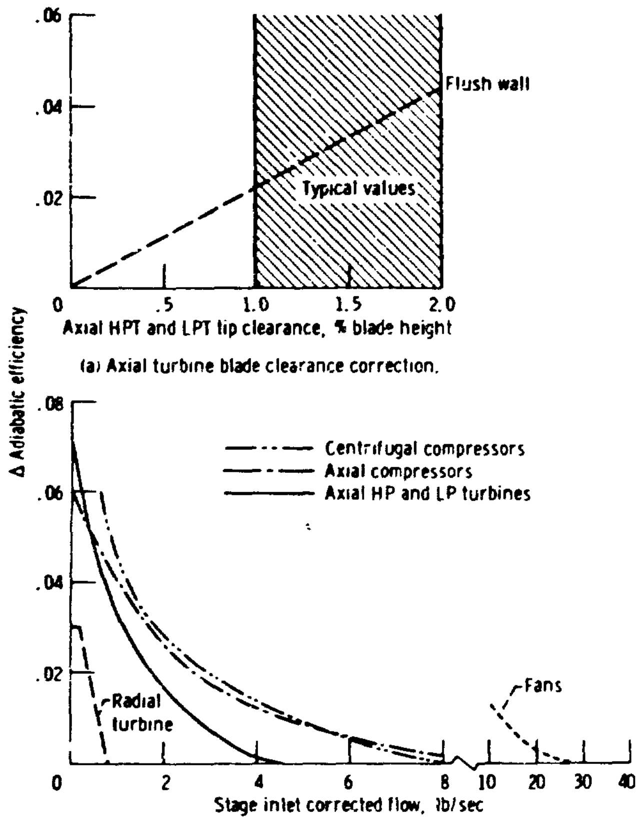
(c) Centrifugal compressor stage adiabatic efficiency.

Figure C1. - Compressor efficiency variation with pressure ratio and technology. Note: For multistage axial compressor, assume equal stage P/P, for 2-stage centrifugal, assume 60/40 P/P split and add 2% ΔP loss between stages.



(b) Radial.

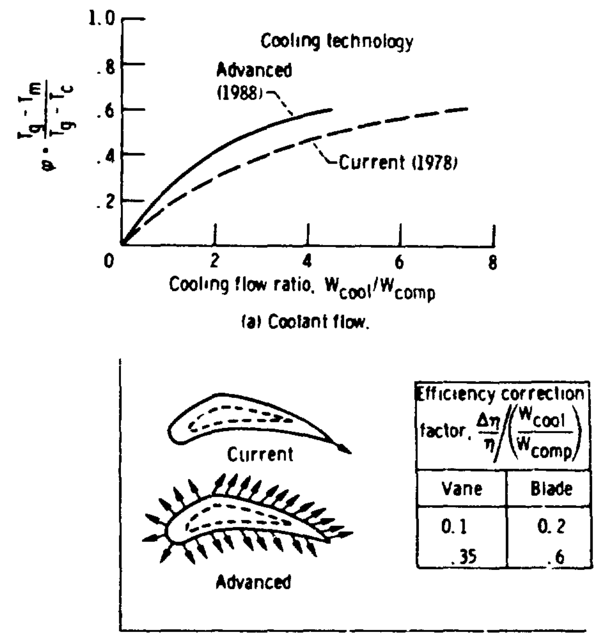
Figure C2. - Turbine efficiency variation with stage work factor and technology. Note. For a radial turbine a stage work factor below 0.9 is not utilized; instead V_{tip} is reduced to maintain maximum η at 0.9 work factor.



(a) Axial turbine blade clearance correction.

(b) Component size correction.

Figure C3. - Turbine and compressor efficiency correction factors.



(a) Coolant flow.

(b) Cooling configuration.

Figure C4. - Turbine stage cooling requirements and associated efficiency correction factor.

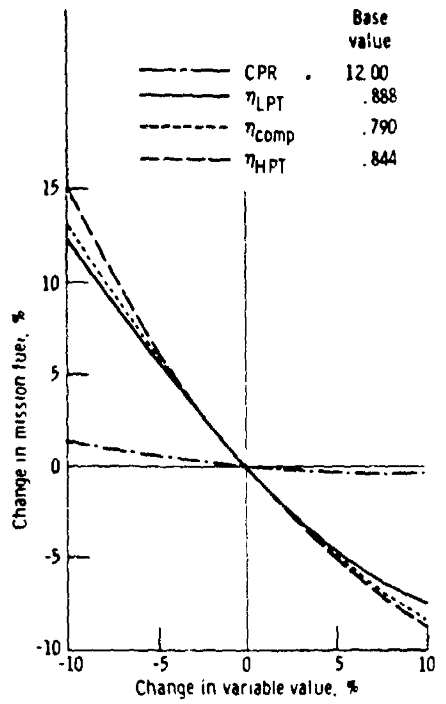


Figure D1. - Mission fuel changes with changes in several parameters. 6 Passenger light twin turboprop.

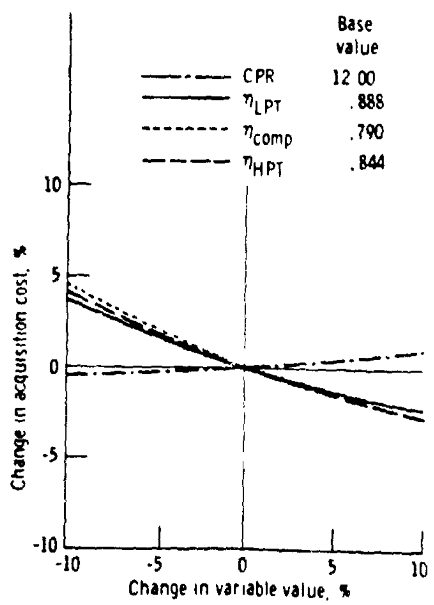


Figure D2. - Acquisition cost changes with changes in several parameters. 6 Passenger light twin turboprop.

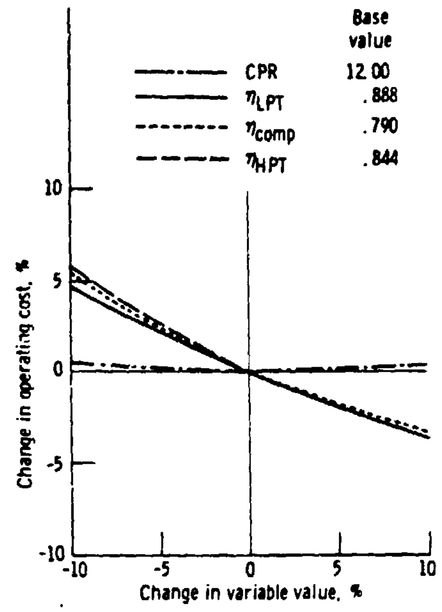


Figure D3. - Operating cost changes with changes in several parameters. 6 Passenger light twin turboprop

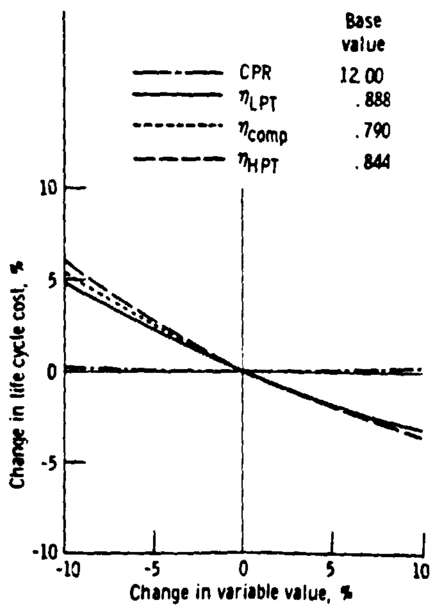


Figure D4. - Life cycle cost changes with changes in several parameters. 6 Passenger light twin turboprop.

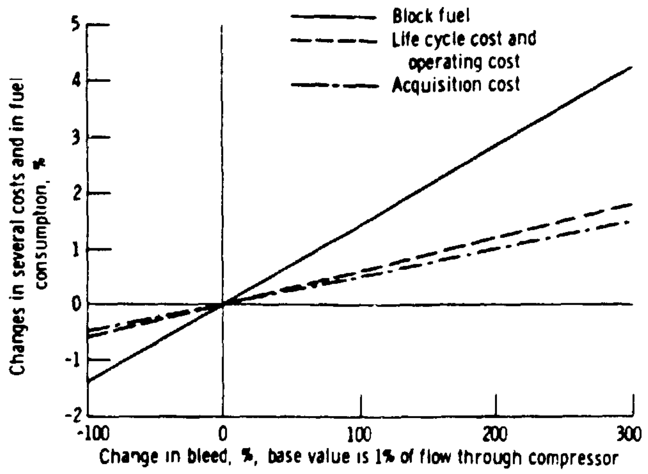


Figure D5. - Changes in mission fuel, acquisition cost, operating cost, and life cycle cost with changes in engine bleed. 6 Passenger light twin turboprop.

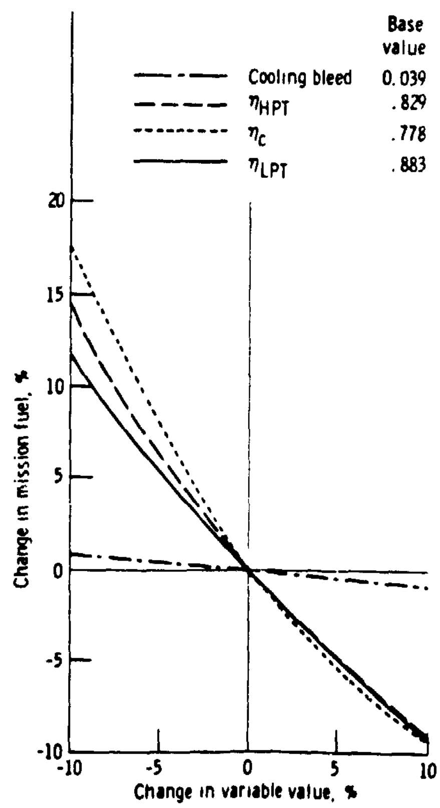


Figure D6. - Mission fuel changes with changes in several parameters. Light single engine helicopter.

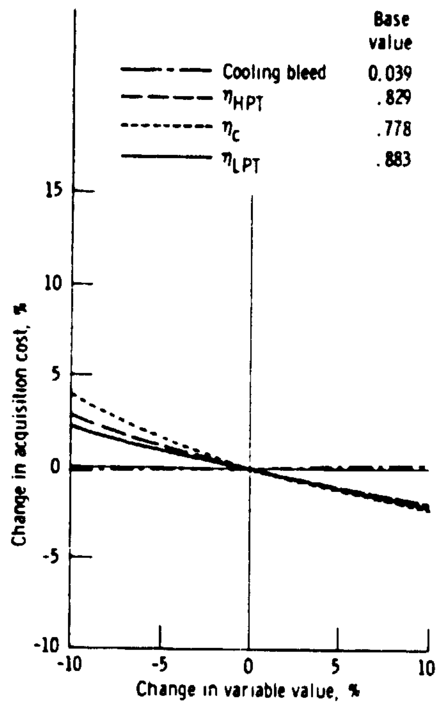


Figure D7. - Acquisition cost changes with changes in several parameters. Light single engine helicopter.

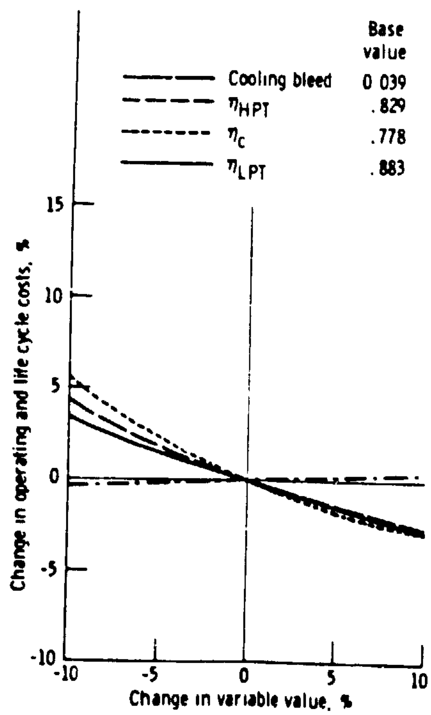


Figure D8. - Operating cost and life cycle cost changes with changes in several parameters Light single engine helicopter

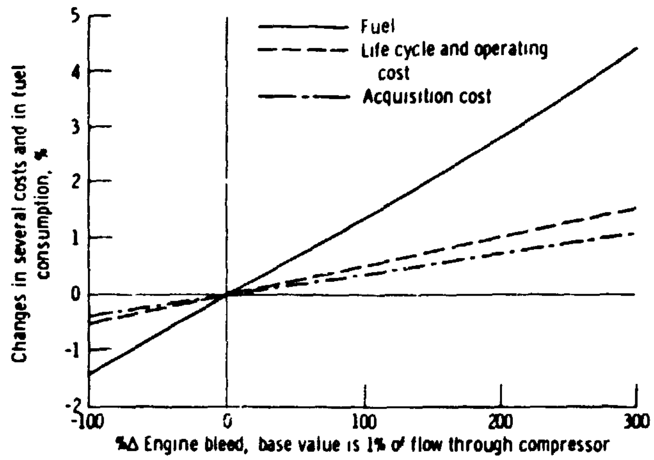


Figure D9. - Changes in mission fuel, acquisition cost, operating cost, and life cycle cost with changes in engine bleed. Light single engine helicopter.

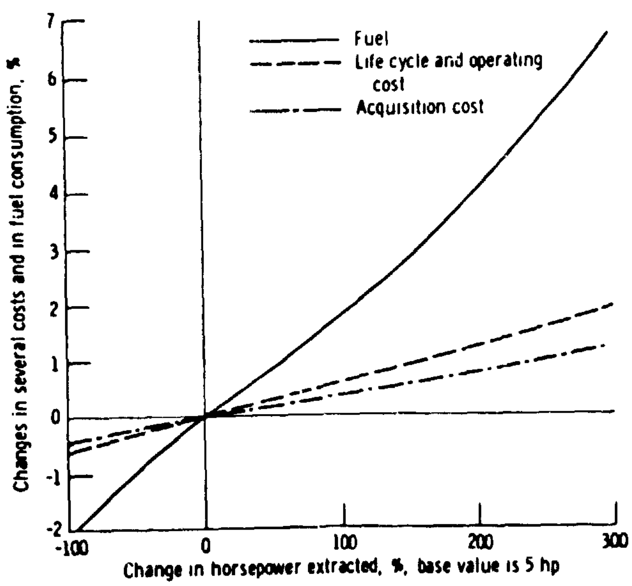


Figure D10. - Changes in mission fuel, acquisition cost, operating cost, and life cycle cost with changes in horsepower extraction. Light single engine helicopter.