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REGENERATIVE ENGINE POWERED AIRCRAFT DESIGN STUDY

By

Richard D. Semple

January 1972

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
FORT EUSTIS, VIRGINIA

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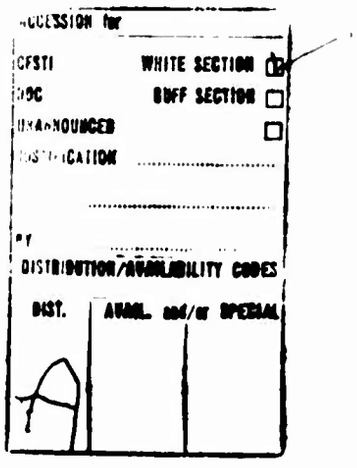
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13 ABSTRACT

Characteristics of aircraft with integrated regenerative engine propulsion systems and aircraft with nonregenerative turboshaft engines were determined in order to assess the relative advantages and disadvantages of the engines in a utility transport helicopter application.

Conceptual designs of aircraft, powered by regenerative engines and nonregenerative engines of approximately 1000 shp, were based on existing and future Army mission requirements for a typical utility transport helicopter. Comparative weight and performance parameters, reliability aspects, maintenance requirements, life-cycle cost and cost effectiveness for various missions, and overall mission effectiveness and system cost were assessed for these aircraft.

The aircraft powered by a 0.65 effectiveness regenerative engine had the lowest take-off gross weight, but it was only 2 percent lighter than its counterpart with a non-regenerative engine. The range or endurance requirements postulated for the various missions of the utility helicopter were not long enough to realize large savings in fuel, and so the differences in life-cycle cost and cost effectiveness, although small, favored the aircraft with a simple-cycle engine.

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	ROLE	WT	ROLE	WT	ROLE	WT
Regenerative Engines						
Nonregenerative Engines						
Utility Transport Helicopter						
Takeoff Gross Weight						
Reliability						
Maintenance Requirements						
Life-Cycle Cost						

DEPARTMENT OF THE ARMY
U.S ARMY AIR MOBILITY RESEARCH & DEVELOPMENT LABORATORY
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FORT EUSTIS, VIRGINIA 23604

This report was prepared by the Boeing Company, Vertol Division, under the terms of Contract DAAJ02-70-C-0061. It describes the results of a study to compare the advantages and disadvantages (using a realistic mission) of utility size helicopters using advanced simple-cycle and advanced integral regenerative gas turbine engines of the 1000-hp class.

The object of this contractual effort was to conduct conceptual designs of aircraft powered by optimum advanced technology nonregenerative gas turbine engines and integral regenerative gas turbines and, subsequently, to perform comparative analyses and evaluations relative to weight and performance parameters, reliability and maintainability factors, life-cycle cost, and cost effectiveness for various missions.

In general, the aircraft with the advanced technology simple-cycle engine had the lowest life-cycle cost and the optimum cost effectiveness. The aircraft powered by a 0.65 effectiveness regenerative engine had the lowest takeoff weights; however, the differences among the various aircraft were relatively small--approximately 2% in gross weight and 1% in cost effectiveness. Thus the engines are considered to be a standoff for the study utility mission.

The technical manager for this contract was Mr. James Gomez, Propulsion Division.

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REGENERATIVE ENGINE POWERED AIRCRAFT DESIGN STUDY

Final Report

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By

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Prepared by

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for

EUSTIS DIRECTORATE
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FORT EUSTIS, VIRGINIA

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ABSTRACT

This report describes the results of a study to determine the characteristics of aircraft with integrated regenerative engine propulsion systems and aircraft with nonregenerative turboshaft engines, and to assess the advantages and disadvantages of regenerative engines compared to nonregenerative engines in a utility transport helicopter application.

Conceptual designs of aircraft, powered by regenerative engines and nonregenerative engines of approximately 1000 shp, were based on existing and future Army mission requirements for a utility transport helicopter. Comparative analyses and evaluations were made of the utility helicopter configurations with optimum integrated propulsion systems, including:

- A nonregenerative turboshaft engine incorporating advanced technology
- A nonregenerative turboshaft engine incorporating available technology
- Three advanced-technology regenerative engines with different values of regenerator effectiveness

Comparative weight and performance parameters, reliability aspects, maintenance requirements, life-cycle cost and cost effectiveness for various missions, and overall mission effectiveness and system cost were assessed for aircraft with regenerative and nonregenerative engine propulsion systems.

The aircraft powered by a 0.65 effectiveness regenerative engine had the lowest takeoff gross weight and empty weight. However, compared with an optimum nonregenerative engine design utilizing advanced technologies, the improvements in specific fuel consumption which could be achieved with the regenerative engine were only 15 to 17 percent. Consequently, the takeoff gross weight of the aircraft with the 0.65 effectiveness regenerative engine was only 2 percent lighter than its counterpart with a nonregenerative engine. The range or endurance postulated for the various missions of the utility helicopter was not long enough to realize large savings in fuel, and so the differences in life-cycle cost and cost effectiveness, although small, favored the aircraft with a simple-cycle engine. A regenerative engine powered aircraft would be superior for range requirements greater than those of the design mission, however.

FORLWORD

This report completes the conceptual design study of aircraft powered by regenerative and nonregenerative engines authorized by Contract DAAJ02-70-C-0061, Regenerative Engine Powered Aircraft Design Study.

The authors acknowledge the assistance of AiResearch Manufacturing Division, The Garrett Corporation, Phoenix, Arizona. In a previous study under Contract DAAJ02-69-C-0087, they designed lightweight integral regenerative gas turbines which were the basis for this contractual effort. During the course of the aircraft design study, AiResearch provided reliability and maintainability data for advanced-technology regenerative and nonregenerative turboshaft engines. They were also helpful in providing additional technical data relative to engine performance and cost.

We also [illegible] our [illegible] to the General Electric Company, Military Engine Division, Lynn, Massachusetts, which played an important role in defining the characteristics of available-technology turboshaft engines.

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LIST OF SYMBOLS

A	circular area swept by rotor blades, ft^2
A_W	wetted area (for drag calculation), ft^2
b	number of blades, main rotor
b_{TR}	number of blades, tail rotor
c	rotor blade chord, ft
C_D	drag coefficient, $\frac{\text{Drag}}{\rho F_e \frac{V^2}{2}}$
C_F	coefficient of friction
C_T	rotor thrust coefficient, $\frac{T_R/A}{\rho V_T^2}$, dimensionless
C_{TR}	tail rotor blade chord, ft
D	rotor diameter, ft
DGW	aircraft design gross weight, lb
DL	rotor disc loading, lb/ft^2
EW	aircraft empty weight, lb
F_e	equivalent flat-plate area (for drag calculation), ft^2
FH	flight hours, hr
FM	rotor figure of merit, $\frac{\left(\frac{T_R}{DGW}\right)^{3/2}_{DGW} \sqrt{\frac{DGW}{A}}}{550 \sqrt{2\rho}\ \ RHP}$
GW	aircraft gross weight, lb
GW_O	gross weight at start of selected mission segment, lb
g	acceleration due to gravity, ft/sec^2
HP_{TR}	tail rotor power, hp
HP_X	transmission power rating, hp

h	distance from base of vertical tail to [illegible] of tail rotor, ft
J_R	trend constant for rotor group weight
k_1, k_2	trend constants for engine dry [illegible]
k_D	droop factor, rotor group [illegible] trend equation
L_c	fuselage length, nose to [illegible] cabin floor, ft
L_{rw}	length of ramp well and [illegible] bay, ft
$\bar{L}$	generalized rotor [illegible] meter, $\frac{GW}{qb^2\sigma}$
M	Mach number
MMH	maintenance man-hours, hr
MRP	engine Military Rated Power (30-minute), shp
MTBF	mean time between failure, hr
MTBR	mean time between removal, hr
MTTR	mean time to repair, hr
N	shaft speed, rpm
N_e	number of engines
N_R	main rotor speed, rpm
NRP	engine Normal Rated Power (con[illegible] rating, shp
OD	engine outside diameter, in.
OEI	one engine inoperative
OHRS	engine operating hours, hr

P	generalized rotor power parameter, $\frac{33{,}000\ RHP}{\rho D^2 \sigma V^3}$
p	pressure, psf
q	dynamic pressure, $\frac{\rho V^2}{2}$
R	main rotor radius, ft
R_{TR}	tail rotor radius, ft
RHP	rotor horsepower, hp
r	distance from center of rotation to point of blade attachment, main rotor, ft
r_{TR}	distance from center of rotation to point of blade attachment, tail rotor, ft
S_f	fuselage wetted area, ft^2
S_H	planform area, horizontal tail, ft^2
S_V	vertical tail area, ft^2
SFC	engine specific fuel consumption, lb/hr/hp
SHP	engine shaft horsepower, shp
SHP*	engine shaft horsepower at Military rating, sea level, 59°F, shp
T	temperature, °R
T_4	turbine-inlet temperature, °F °R
T_R	rotor thrust, lb
TAS	true airspeed, kt
TBO	time between overhaul, hr
TMA_H	horizontal tail moment arm, ft
TMA_V	vertical tail moment arm, ft
TIT	turbine-inlet temperature, °F °R

V	aircraft speed, kn
V_{CR}	aircraft cruise speed, kn
V_{CR_O}	cruise speed at start of selected mission segment, kn
V_{MAX}	maximum speed, kn
V_{OPT}	aircraft cruise speed for maximum range, kn
V_T	rotor blade tip speed, ft/sec
VRC	vertical rate of climb, ft/min
W_{BG}	aircraft body group weight, lb
W_D	drive system weight, lb
W_e	engine dry weight, lb
W_F	weight of mission fuel, lb
W_{FC}	flight controls group weight, lb
W_{FS}	fuel system group weight, lb
W_{HT}	horizontal tail weight, lb
W_{LG}	landing gear group weight, lb
W_{MT}	engine mount weight, lb
W_R	main rotor group weight lb
W_{TR}	tail rotor group weight, lb
W_{VCD}	vibration control devices weight, lb
W_{VT}	vertical tail weight, lb
w_a	engine inlet airflow, lb/sec
w_f	engine fuel flow, lb/hr

ΔCG	allowable travel in aircraft center of gravity, ft
δ	pressure divided by NASA standard day pressure, P/14.696
η	adiabatic efficiency
η_{cr}	crash-load factor for engine mount weight (20 for internal mounts, 12 for external mounts)
η_p	propulsive efficiency of tail rotor
η_T	transmission efficiency
η_u	ultimate load factor for tail/body group weight trends (=1.5 x limit load factor)
θ	temperature ($^{\circ}R$) divided by NASA standard day temperature, T/518.7
ρ	ambient air density, lb/ft^3
σ	rotor solidity, $\frac{RbC}{\pi R^2}$

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am	ambient conditions
L	design limit value
R	main rotor
TR	tail rotor

SUMMARY

The regenerative turboshaft engine uses a heat exchanger to recover much of the heat energy normally lost in the exhaust gases. The heat is transferred to compressor discharge air and reduces the amount of fuel required by the combustor to achieve desired turbine-inlet temperatures. The result is an improvement in the specific fuel consumption (SFC) of the regenerative engine compared to the simple-cycle turboshaft engine.

The term regenerator usually is applied to the rotating, periodic-flow type of heat exchanger, while the stationary heat exchanger is called a recuperator. The terms are used interchangeably in this document. The nonregenerative turboshaft engine, which has a single-spool gas generator and a free power turbine, is called a simple-cycle turboshaft engine.

During the past ten years, development efforts on regenerators and regenerative engines have proven the feasibility of various heat exchanger concepts. Studies have shown potentially significant improvements in the performance of future Army aircraft powered by regenerative engines. In helicopter flight test programs, performance data for existing engines, modified to accommodate a "bolt-on" recuperator, substantiated improvements in fuel requirements and range capability. Reductions in engine exhaust noise and infrared signature accompanied these performance benefits. To properly assess the merits of the regenerative cycle, however, these advantages must be evaluated against increased engine weight, cost, and maintenance requirements.

In a recent program conducted for the U.S. Army under Contract DAAJ02-69-C-0087, analytical and design efforts were directed toward achieving compact, lightweight regenerative engine designs of approximately 1000 shp. The engine configurations utilized an annular recuperator of tubular construction wrapped around the turbomachinery. The recuperator served as the structural backbone of the engine assembly, resulting in a well-integrated regenerative engine concept. Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, subsequently contracted with The Boeing Company, Vertol Division, to develop conceptual designs of utility transport helicopters, using this advanced engine and regenerator data (Reference 1). The program culminated in comparative analyses and evaluations of design, performance,

reliability, maintainability, life-cycle cost, and cost effectiveness of the aircraft systems, to assess the advantages and disadvantages of regenerative turboshaft engines relative to nonregenerative engines in the utility helicopter installation. This document is the final report of the aircraft design study program.

The aircraft conceptual designs were based on existing and future requirements for a utility transport helicopter. War game scenarios for typical army utility tactical transport aircraft were used in the mission analysis. Boeing selected a design mission and secondary missions which typified Army use of the utility helicopter in all intensities of conflict. Altitude and ambient temperature for each flight regime; takeoff, hover, and climb requirements; cruise speed; and payload and range were selected for utility, medical evacuation, and observation missions.

The design-point compressor pressure ratio for the advanced-technology engines in Reference 1 was fixed at 9:1, which was optimum for the regenerative engines. For a simple-cycle engine, however, a higher pressure ratio would produce a better specific power and a decrease in SFC. A pressure ratio of 14:1 was selected as the optimum value for the simple-cycle engine, and performance data were developed for this engine using the advanced component technologies. Design-point component efficiencies, losses, and cooling-air flows for both the advanced technology regenerative and nonregenerative engines were quite optimistic for an engine with an airflow of 5.0 lb/sec, and the resulting shaft horsepower and SFC presented extremely difficult targets. Therefore, Boeing redefined component performance data for a simple-cycle engine utilizing expected technologies, and calculated overall performance which was consistent with correlations of shaft horsepower and SFC for small turboshaft engines under development or proposed for development. Complete off-design performance data were developed for both advanced-technology and available-technology simple-cycle engines and for three advanced-technology regenerative engines with recuperator effectiveness values of 0.40, 0.65, and 0.80. Performance data for these five engines have been plotted in Figure 1, illustrating the improvements in SFC with increasing recuperator effectiveness, as well as the relative optimism of the advanced-technology engine compared to the available-technology simple-cycle engine.

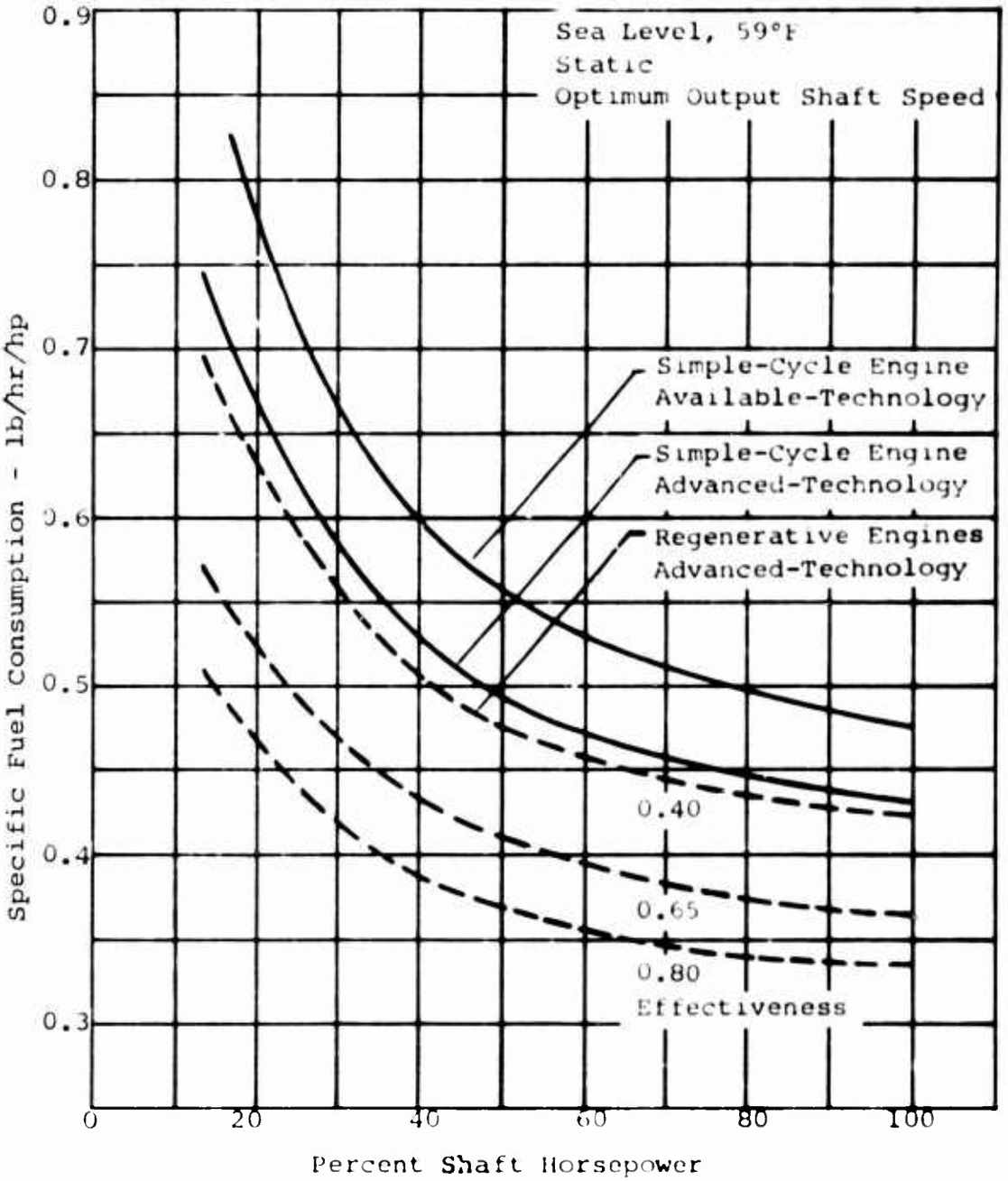


Figure 1. Comparison of Engine Performance for Advanced-Technology Regenerative and Nonregenerative Engines and an Available-Technology Simple-Cycle Engine.

Engine dry weights were calculated for each of these configurations, sized for 1000 shp, and plotted in Figure 2. The increasing slope of the weight curve resulted from the

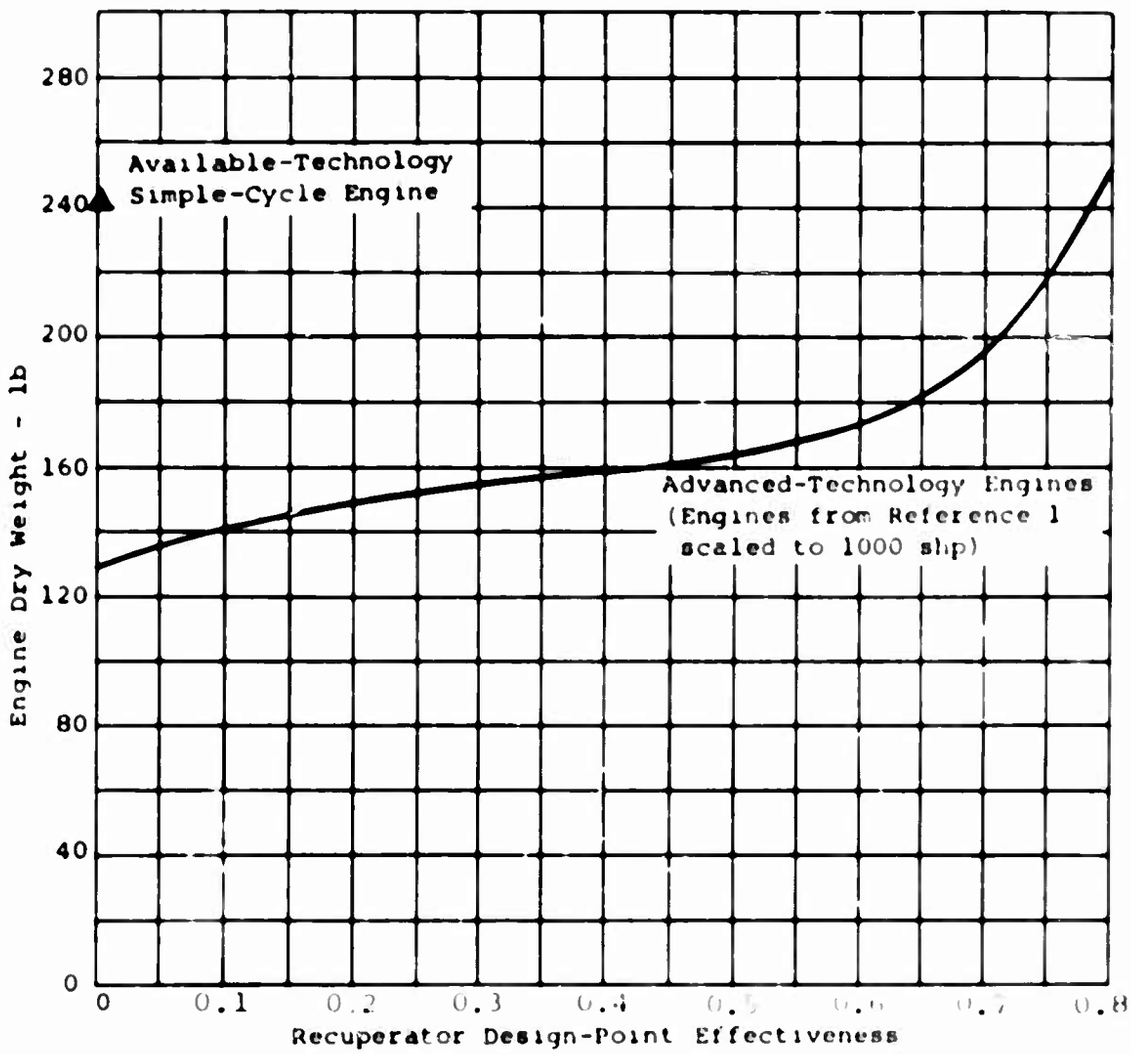


Figure 2. Dry Weights for 1000-SHP Regenerative and Non-regenerative Engines.

rapid increase in recuperator size required to achieve the very high values of effectiveness. This weight increase more than compensated for decreased fuel requirements at high values of effectiveness, at least for the utility mission predicated for the design aircraft. The net result was that 0.65 was approximately the optimum recuperator effectiveness, from the standpoint of aircraft gross weight.

Boeing developed conceptual designs of utility transport helicopters using this regenerative and nonregenerative engine data. The aircraft were based on fixed range and payload requirements for the utility mission. Engines were scaled over a limited range to achieve the helicopter power requirements. Since the scaling range was so limited, engine performance in terms of specific power and SFC was assumed to be the same as that generated for the engines of approximately 1000-shp size, but weight scaling curves were calculated as a function of shaft horsepower. Engine installations were optimized with respect to inlet, exhaust, subsystems, orientation, and other aspects to define an integrated aircraft system with the best possible weight, balance, and drag characteristics. Detailed weight analyses were performed for each aircraft. Takeoff gross weight characteristics have been plotted in Figure 3. The curve verified that 0.65 was approximately the optimum recuperator effectiveness from the standpoint of gross weight, and the aircraft weights were higher at the extremes of the range of recuperator effectiveness values considered. The aircraft with this optimum regenerative engine propulsion system was lighter than one with an advanced-technology nonregenerative engine. The differences among the aircraft weights were very small, however, despite the improvements in SFC which could be achieved with the regenerative engine compared with the optimum advanced-technology simple-cycle engine. Figure 3 also showed a substantial weight difference between aircraft powered by advanced-technology simple-cycle engines and available-technology engines.

In line with the small differences in gross weights among the aircraft powered by advanced-technology regenerative and nonregenerative engines, small differences were also encountered in system cost and cost effectiveness for the various missions as well as overall mission effectiveness and system cost. In the case of cost trends, however, the advanced-technology simple-cycle engine was the optimum powerplant for the typical missions selected for the utility helicopter. Basic differences in development and production costs and maintenance requirements, for the regenerative engines compared to the nonregenerative engine, were too great to be offset by fuel savings.

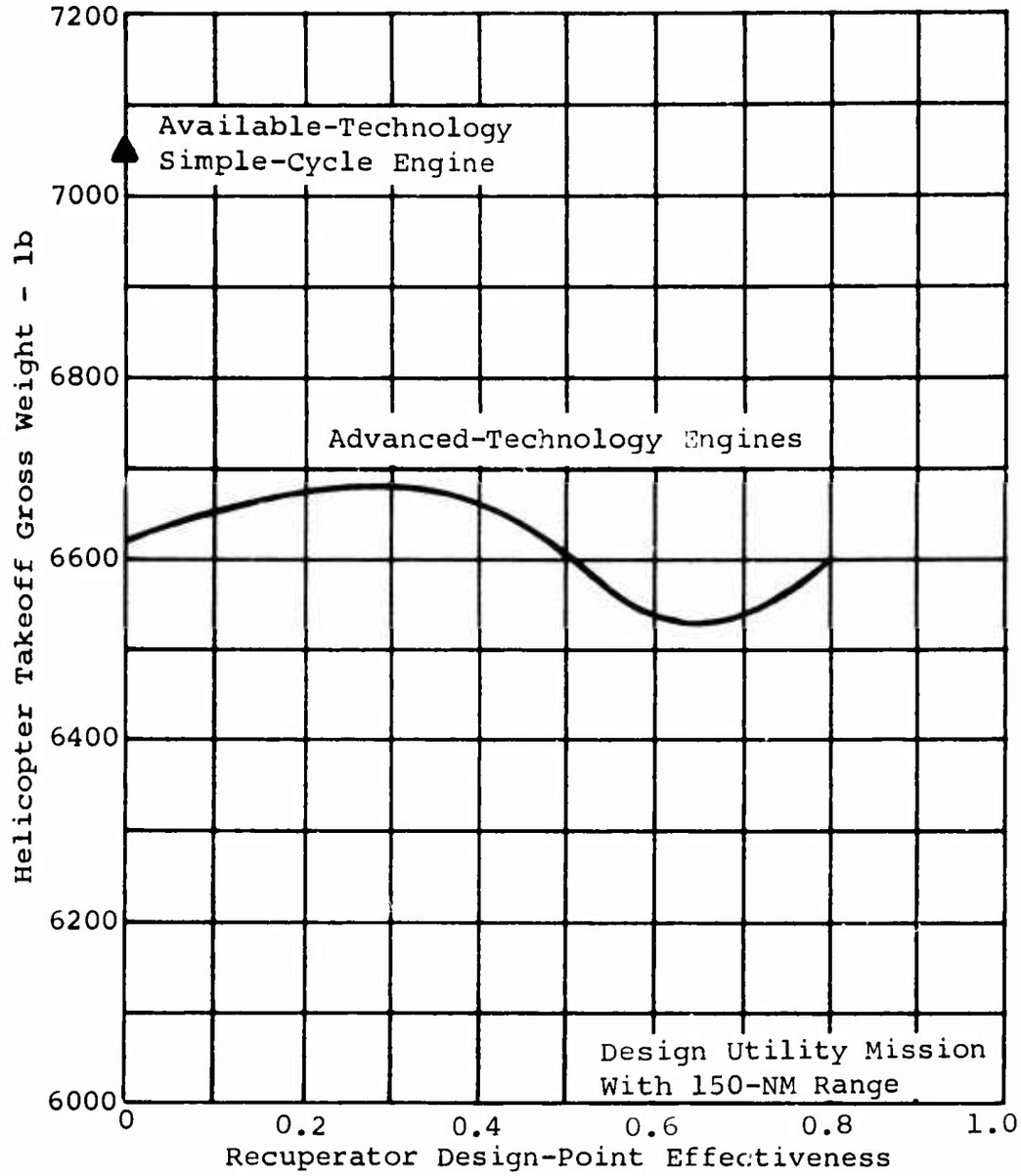


Figure 3. Takeoff Gross Weights for Helicopters With Regenerative and Nonregenerative Engines.

The range or endurance for the various mission roles which typified Army use of the utility helicopter was not sufficiently long to produce appreciable fuel savings with regenerative engines, which would have a significant impact on aircraft weight. A utility mission with a longer range requirement was postulated to investigate the effect of this parameter. The weight trend relationship for fuel tankage was used, with the same fixed tare for plumbing and pumps and

the same percentage of fuel reserves. The gross weights of the aircraft with regenerative and nonregenerative engines, performing the same utility mission described in this report but with a longer range, were plotted in Figure 4. The result was a much greater variation in gross weight among the different aircraft. With this extended range requirement, although there was a greater variation in gross weight, little change was encountered in the trends of life-cycle costs. The aircraft with the advanced-technology simple-cycle engine was still slightly superior in cost to the regenerative-engine powered helicopters (Figure 5).

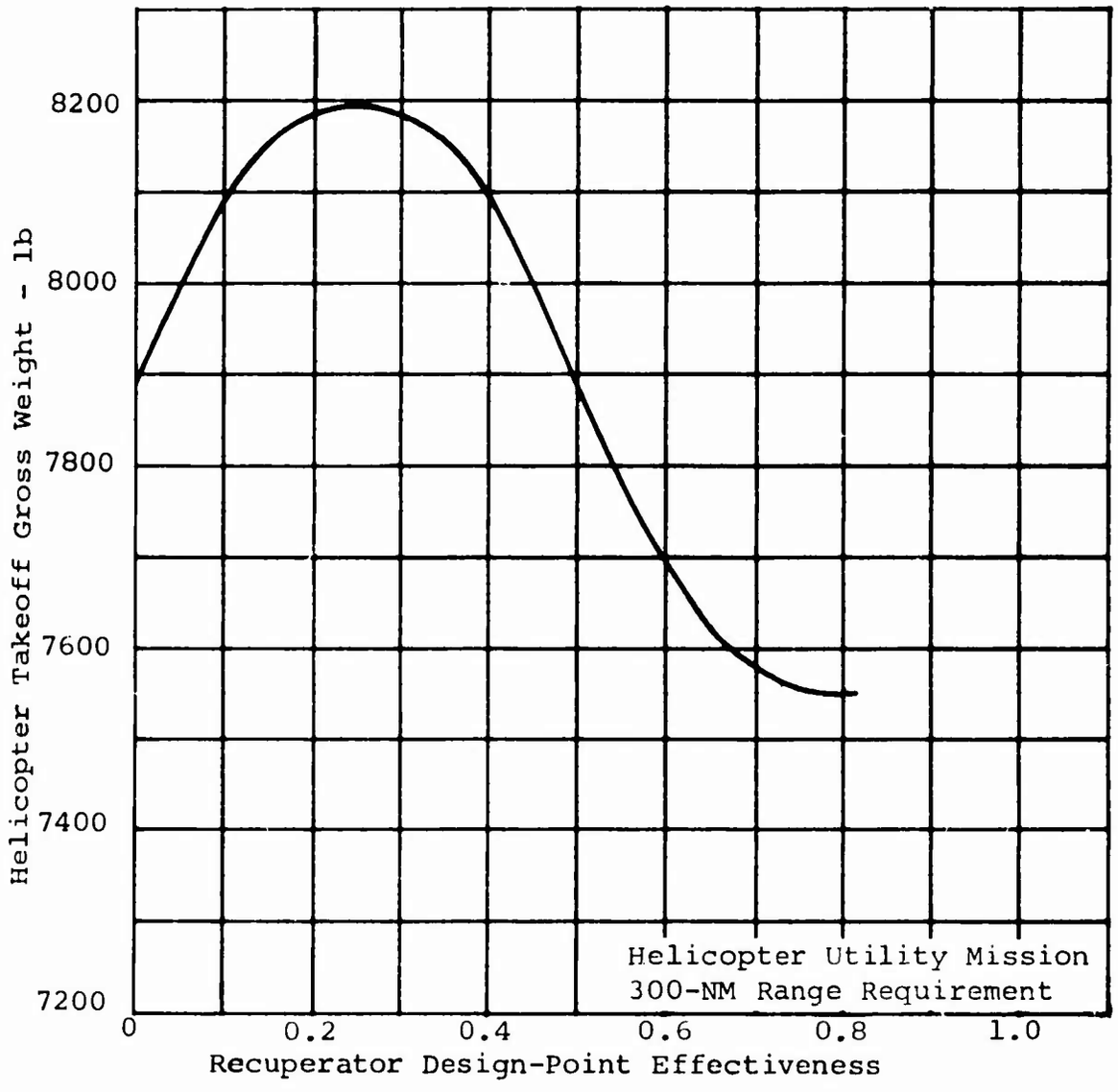


Figure 4. Takeoff Gross Weights for Helicopters With Advanced-Technology Regenerative and Nonregenerative Engines.

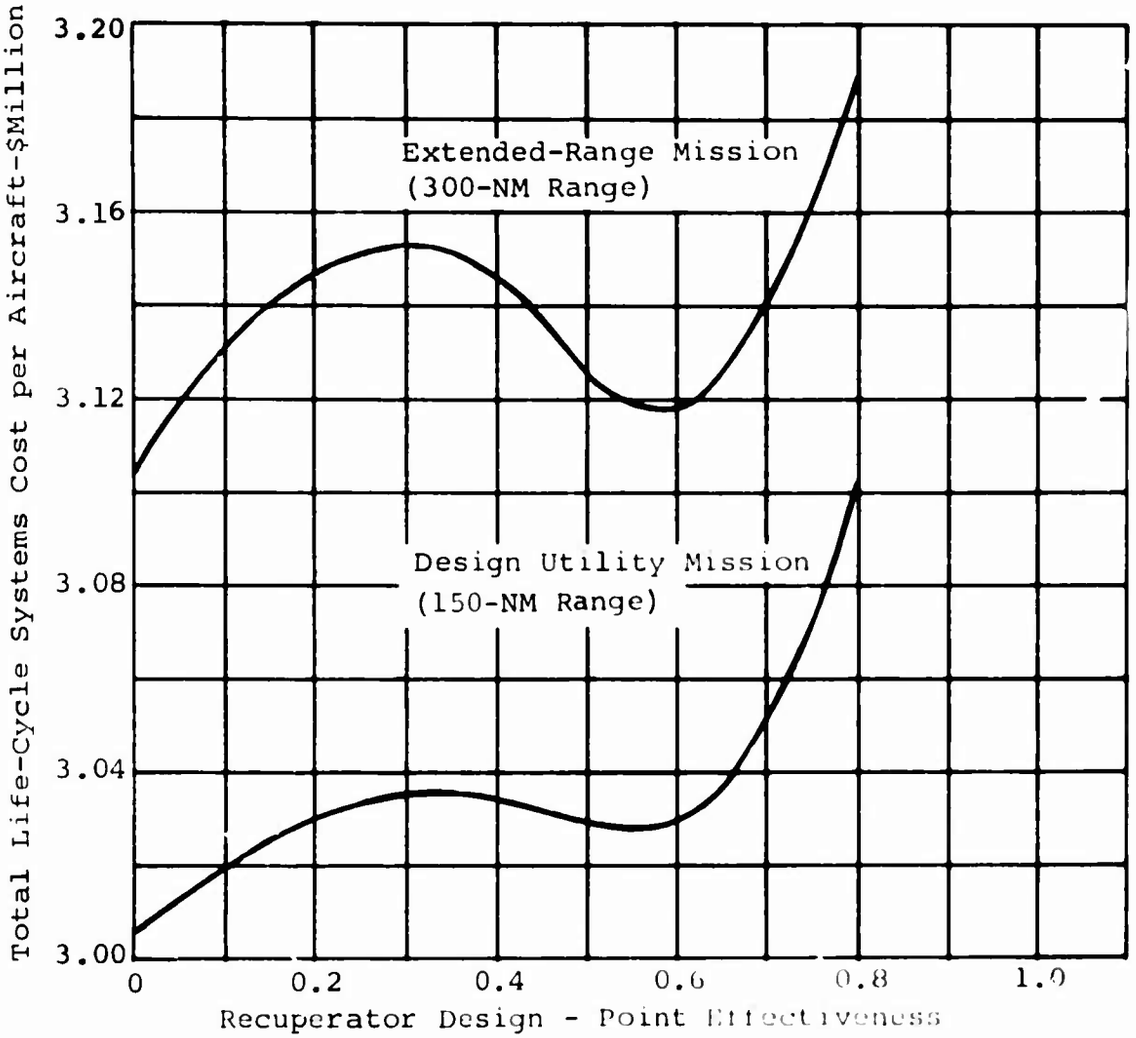


Figure 5. Life-Cycle Costs for Helicopters With Regenerative and Nonregenerative Engines - Design Mission and Extended-Range Mission.

Despite the improvements in SFC which could be achieved with a regenerative engine, compared with an optimum nonregenerative engine design utilizing advanced technologies, large fuel savings would not be realized in a utility helicopter application for the typical Army mission. However, there are other aspects of the regenerative engine installation which could be exploited. These include the reduced infrared signature, improved vulnerability characteristics, or reduced noise - factors which could not be included within the scope of the present program.

INTRODUCTION

The simple-cycle turboshaft engine, with just a single-spool gas generator and a free power turbine, typically dissipates a large proportion of the input fuel energy as exhaust heat. The addition of a heat exchanger between the engine exhaust gas and the compressor exit air improves the thermal efficiency of the engine by recovering some of this heat energy normally lost in the exhaust. The heat exchanger preheats the air entering the burner, reduces the amount of fuel required to reach desired turbine-inlet temperatures, and results in a decrease in specific fuel consumption (SFC).

During the past ten years, development efforts on regenerators and regenerative gas turbine engines have proven the feasibility of various heat exchanger concepts. Concurrently, studies of future Army aircraft powered by regenerative engines have shown promise of significant improvement in their performance. In helicopter flight test programs, regenerative engine performance data for minimum modification conversions of existing engines substantiated predicted improvements in fuel requirements and range capability. Reductions in engine exhaust noise and infrared signature accompanied the performance benefits. These advantages must be compared with increases in engine weight, development and procurement costs, and maintenance requirements to properly assess the merits of the regenerative cycle. The term regenerator usually is applied to the rotating periodic-flow type of heat exchanger, while the stationary heat exchanger is termed a recuperator. In this report, the terms recuperator and regenerator are used interchangeably. In either case, the engine is called a regenerative engine in the following discussions.

The design-point SFC of the regenerative engine is lower than that of the simple-cycle engine, but even more significant than the improvement in design-point performance is the further improvement in SFC at part-power conditions, where the engine operates much of the time in the helicopter installation. Figure 6 demonstrates the benefits of the regenerative engine - its better design-point performance as well as its flatter part-power SFC characteristic.

To date, regenerative engines evaluated in static or flight test programs have been existing engines which were modified to accommodate a "bolt-on" recuperator. Although they performed satisfactorily and demonstrated the structural integrity of the

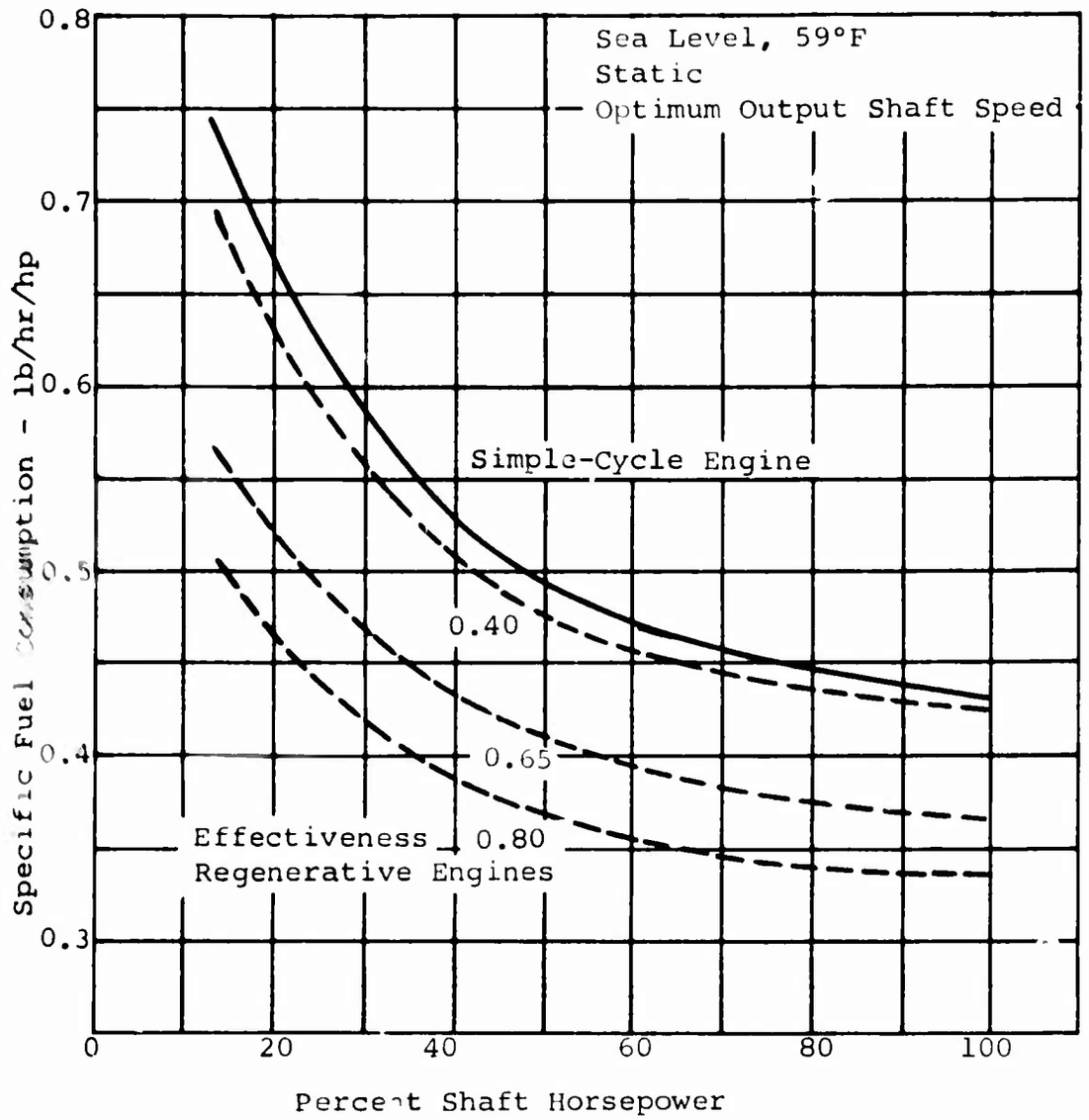


Figure 6. Comparison of Engine Performance for Advanced-Technology Regenerative and Nonregenerative Engines.

heat exchanger, the engines were not an optimum design from the standpoint of performance and weight. In a recent program conducted by AiResearch Manufacturing Company for the U. S. Army, however, analytical and design effort was directed toward achieving integrated regenerative engine designs which were compact and lightweight. The results of this program are presented in Reference 1. The engine concepts utilized an annular recuperator of tubular construction wrapped around the turbomachinery, the recuperator acting as the structural

backbone of the engine assembly. Engine configurations were defined for external installation on the aircraft and for internal installation, with different values of recuperator effectiveness varying from 0.40 to 0.80, and different pressure losses. For comparison purposes, a nonregenerative engine was shown in Reference 1 for the same cycle conditions, although this engine was not used in the aircraft study because it did not lead to optimum simple-cycle performance.

The AiResearch engines represent advanced technologies, with a 2300°F turbine-inlet temperature at the design point and a compressor pressure ratio of 9:1. At the design airflow rate of 5.0 lb/sec, the component efficiencies and cooling-air flows are optimistic and present difficult design targets, which is true also of the resulting power and SFC. This optimism is apparent too in the light weight of the engines. The combination of design-point cycle parameters results in an engine which is sized for approximately 1000 shp. The engines are rear-drive configurations. Performance, weight, dimensional, and cost data are provided for these advanced-technology engines in Reference 1 and are summarized in Table I. Reliability and maintainability data were subsequently developed by AiResearch.

Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, contracted with The Boeing Company, Vertol Division, to conduct a 7-month program to determine the characteristics of aircraft designs with integrated regenerative engine propulsion systems and aircraft with nonregenerative simple-cycle turboshaft engines, and to assess the advantages and disadvantages of regenerative engines compared to nonregenerative engines in a utility transport helicopter application. The helicopter designs were to be accomplished using the advanced engine and regenerator data presented in Reference 1, for engines of approximately 1000 shp. Because of the degree of optimism in component performance, the engine characteristics for simple-cycle and regenerative engines developed by AiResearch were assumed to represent advanced-technology engines which would require substantial development effort to reach production in 1975. To provide a standard of comparison for the advanced-technology engines, Boeing defined similar characteristics for a simple-cycle engine utilizing available technologies - component performance, losses, and weights typical of engines under development or proposed for development. Therefore, conceptual designs of a utility transport helicopter were developed with

TABLE I. ADVANCED-TECHNOLOGY ENGINE CHARACTERISTICS, DESIGN POINT (REFERENCE 1)

Engine Configuration				
Compressor Pressure Ratio	9:1			
Turbine-Inlet Temperature, °F	2300			
Engine Configuration -				
Compressor	1 Axial Stage plus 1 Centrifugal			
Gas Generator Turbine	1 Axial Stage			
Power Turbine	2 Axial Stages			
	Regenerative Engines			Simple-Cycle Engine
Engine Airflow - lb/sec	5.0	5.0	5.0	4.71
Recuperator Effectiveness	.40	.65	.80	-
Recuperator Pressure Loss, Total Air/Gas Side, %	10.0	6.0	4.0	-
Specific Horsepower, shp/lb/sec	187.5	192.54	194.70	204.4
SFC, lb/hr/hp	.425	.365	.353	.366
Shaft Horsepower, shp	937.5	962.7	973.5	962.7
External Mounting (A)				
Engine Weight, lb	161.1	202.8	303.7	110.0
Power/Weight, shp/lb	5.82	4.75	3.20	8.75
Engine OD, in.	19.60	23.45	26.3	14.0
Engine Length, in.	28.7	29.9	40.5	28.55
Internal Mounting (B)				
Engine Weight, lb	147.7	173.6	248.3	-
Power/Weight, shp/lb	6.35	5.55	3.92	-
Engine OD, in.	19.58	19.04	23.18	-
Engine Length, in.	28.5	28.5	33.0	-

Boeing-defined advanced-technology and available-technology simple-cycle engines and with advanced-technology regenerative engines having recuperators with different values of effectiveness as defined in Reference 1. The aircraft were based on fixed range and payload requirements for the utility

mission, and consequently the engines were scaled to achieve helicopter power requirements. Because the scaling range was small, no changes in engine performance were assumed, but weight scaling curves were generated as a function of shaft horsepower from the baseline engines of approximately 1000 shp.

Twin engines in the aircraft configuration would enhance mission reliability and offer a desirable asset for the utility helicopter operating in a battlefield environment. Engine inoperative requirements were determined, and twin-engine aircraft (for the same mission and payload) were included among the alternative conceptual designs. With the exception of increased mission reliability, however, all other factors favored single-engine installations compared with twin engines. Because this program was concerned primarily with tradeoffs between regenerative and nonregenerative engines, twin-engine aircraft were eliminated from the study.

Weight and performance parameters, reliability aspects and maintenance requirements, and life-cycle cost and cost effectiveness were assessed for the single-engine aircraft powered by regenerative and nonregenerative engines of approximately 1000 shp. For reference purposes, analyses and data pertaining to twin-engine aircraft and the smaller engines were assembled in Appendix I. This document is the final report of the study program.

The program was organized into five work tasks, as follows:

Task 1 - Turboshaft Engine Design Parameters
Task 2 - Regenerative Engine Design Parameters
Task 3 - Aircraft Configuration Studies
Task 4 - Propulsion/Airframe Integration
Task 5 - Aircraft Comparative Analyses

Each work task is described in a separate section of this report. The first section is devoted to definition of the design mission and secondary missions selected to typify Army use of the helicopter. Program conclusions are presented in the last section of this report.

AIRCRAFT MISSION DEFINITION

The aircraft conceptual designs were based on existing and future mission requirements for a utility transport helicopter, to insure that the study results would be meaningful. Toward this end, Boeing selected a design mission and secondary missions which typify Army use of the utility helicopter in all intensities of conflict, with a frequency distribution for appropriate weighting of the defined missions. Mission selection included altitude and ambient temperature for each flight regime; takeoff, hover, and climb requirements; cruise speed; and payload and range.

The War Game Scenarios for the Army's Utility Tactical Transport Aircraft System (UTTAS), published by the U. S. Army Combat Developments Command, were selected as the basis of the mission analysis (References 2,3,4, and 5). Most current Army planning visualizes mid- and high-intensity conflict as the basis for equipping and organizing the Army of the future. Although considerable mission data for low-intensity conflict are available from Vietnam experience, they are not directly applicable to higher intensities of conflict, while the USACDC (U.S. Army Combat Developments Command) UTTAS Study Scenarios address all intensities of conflict. The scenarios were one of the principal means of establishing the mission and performance envelope for the UTTAS utility transport helicopter. They include missions currently performed by utility aircraft and missions visualized for the future, possibly with new equipment and/or new organization, and are appropriate to use in a study relating to a future regenerative engine. In the scenarios were 983 tasks itemized for helicopter accomplishment in the low-intensity conflict, 793 tasks in the mid-intensity conflict, and 278 in the high-intensity conflict. Of the total tasks, those that fall into the following categories were determined (Table II):

- Utility (defined as search and destroy, command and control, civil affairs, psychological warfare, and POW extraction)
- Medical evacuation (including air crash rescue and air ambulance)
- Observation

These three categories represented 65 percent of the total

tasks for helicopter accomplishment in the low-intensity conflict, 43 percent in the mid-intensity conflict, and 26 percent in the high-intensity conflict. The decrease in the percentage of tasks to be accomplished with the increase in level of conflict was indicative of the greater demand made of helicopters for large troop movements, equipment and artillery movements, and resupply movements as the intensity of conflict increased.

TABLE II. HELICOPTER TASKS IN VARIOUS LEVELS OF CONFLICT

	Level of Conflict					
	Low Intensity		Mid Intensity		High Intensity	
	No. of Tasks	%	No. of Tasks	%	No. of Tasks	%
Utility	389	61	179	52	43	61
Medical Evacuation	182	28	152	44	16	22
Observation	68	11	12	4	12	17
Total	639	100	343	100	71	100

South Vietnam was considered representative of low-intensity, South Korea representative of mid-intensity, and the Federal Republic of Germany representative of high-intensity conflict. The scenarios were analyzed to determine altitude-temperature conditions for the aircraft. To satisfy the requirements for a mid-intensity conflict, the following capabilities were defined for the utility helicopter:

- Takeoff power criteria of 4000 feet, 95°F ambient conditions with a 500-fpm vertical rate of climb at 95 percent Military Power
- Cruise Speed from 110 to 140 knots
- Payload and range as described in the mission profiles in subsequent paragraphs.

A helicopter with this capability should perform the mission for which it was designed 96 percent of the time in South Korea and 98 percent of the time in the Federal Republic of

Germany. A twin-engine configuration could enhance mission reliability, and two engines would be a desirable asset for the utility helicopter operating in a battlefield environment. Accordingly, engine inoperative requirements were determined, and twin-engine aircraft were included among the alternative conceptual designs.

DESIGN MISSION

In the design (utility) mission, an airmobile operation will be conducted using a battalion combat team as the assault force (Figure 7). The battalion commander elects to use the utility helicopter for command and control. The aviation unit commander, the air liaison officer, and the artillery observer accompany him in the aircraft to provide control of supporting elements.

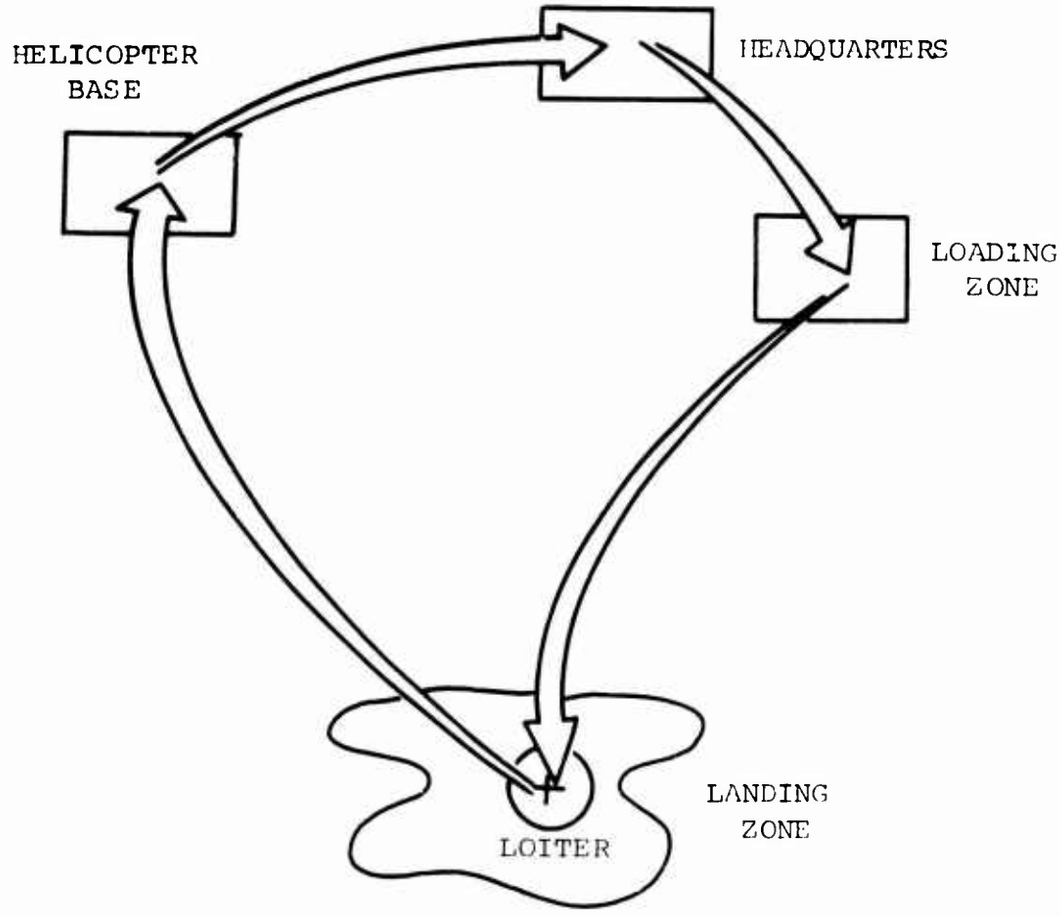


Figure 7. Helicopter Utility Mission.

The helicopter transports three passengers from the helicopter base to battalion headquarters to pick up the battalion commander. From here they fly to the loading zone to supervise the loading of the aircraft. The helicopter then transports the four passengers to the landing zone. The helicopter loiters for 20 minutes to observe preparatory artillery and close air support during the landing and departure of the transport helicopters, and to observe and direct the positioning of maneuver elements and the coordination of supporting air and artillery fire. The helicopter then lands and discharges all passengers except the aviation unit commander, who returns with the helicopter to the base.

The mission demands a payload of four passengers; 2 to 3 hours endurance, including takeoffs and landings; or a maximum range of 150 nautical miles with a 20-minute loiter, and 4 takeoffs and landings. Figure 8 presents the cumulative frequency of occurrence of the distance of movement for the utility mission, from the USACDC UTTAS scenarios. The typical mission profile follows:

1. Warm up 2 minutes at Normal Rated Power.
2. Take off and hover out of ground effect, 2 minutes.
3. Cruise outbound.
4. Loiter 20 minutes at 75 nautical miles.
5. Land, unload, take off, and hover out of ground effect, 2 minutes.
6. Cruise inbound 75 nautical miles.
7. Land with 10 percent of initial fuel.

This mission profile does not specify explicitly the two intermediate landings and takeoffs. They are accounted for in the hover time. The intermediate cruise distances are not defined more precisely to simplify the design of the helicopter.

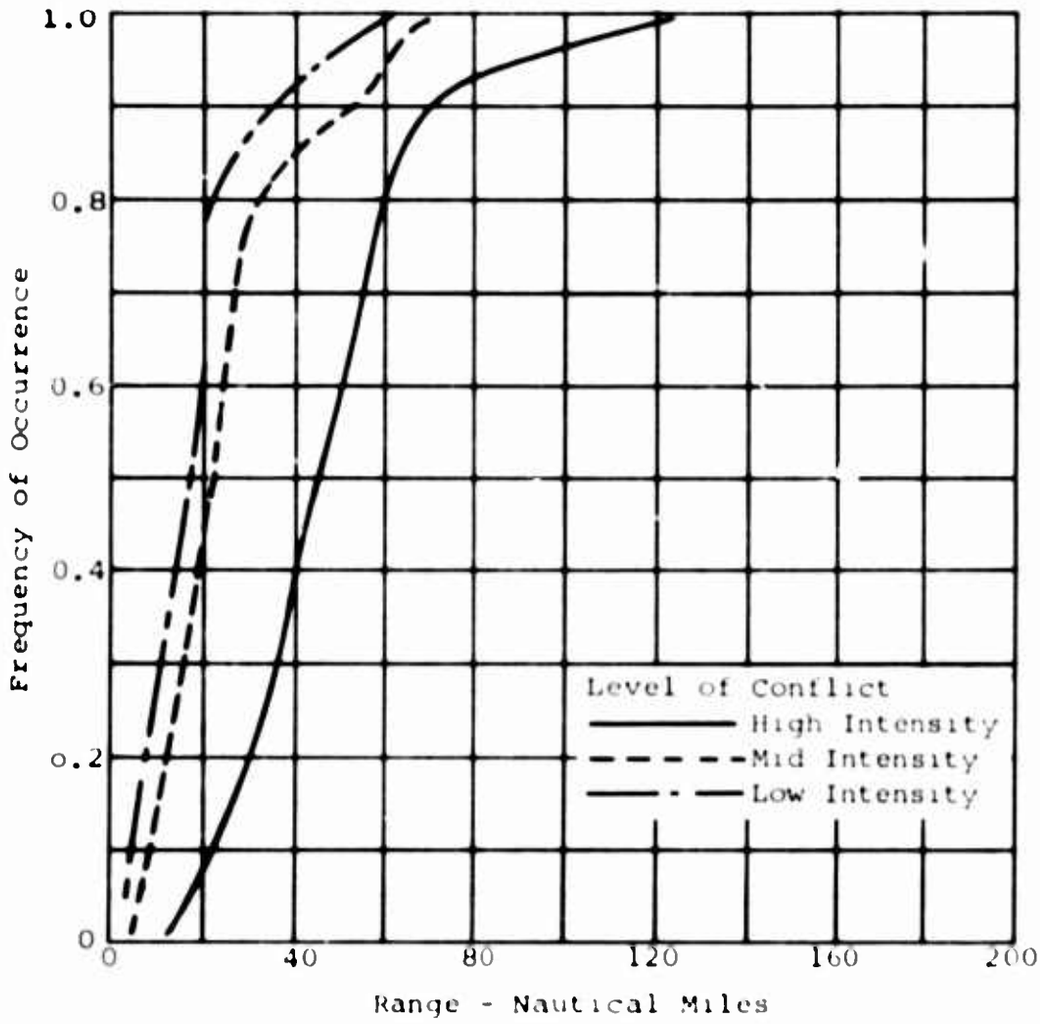


Figure 8. Cumulative Frequency of Occurrence of Distance of Movement for Utility Mission.

SECONDARY MISSIONS

The typical scenario for the medical evacuation mission (Figure 9) requires evacuation of 12 disabled wounded personnel from a forward area to a field hospital 15 nautical miles to the rear.

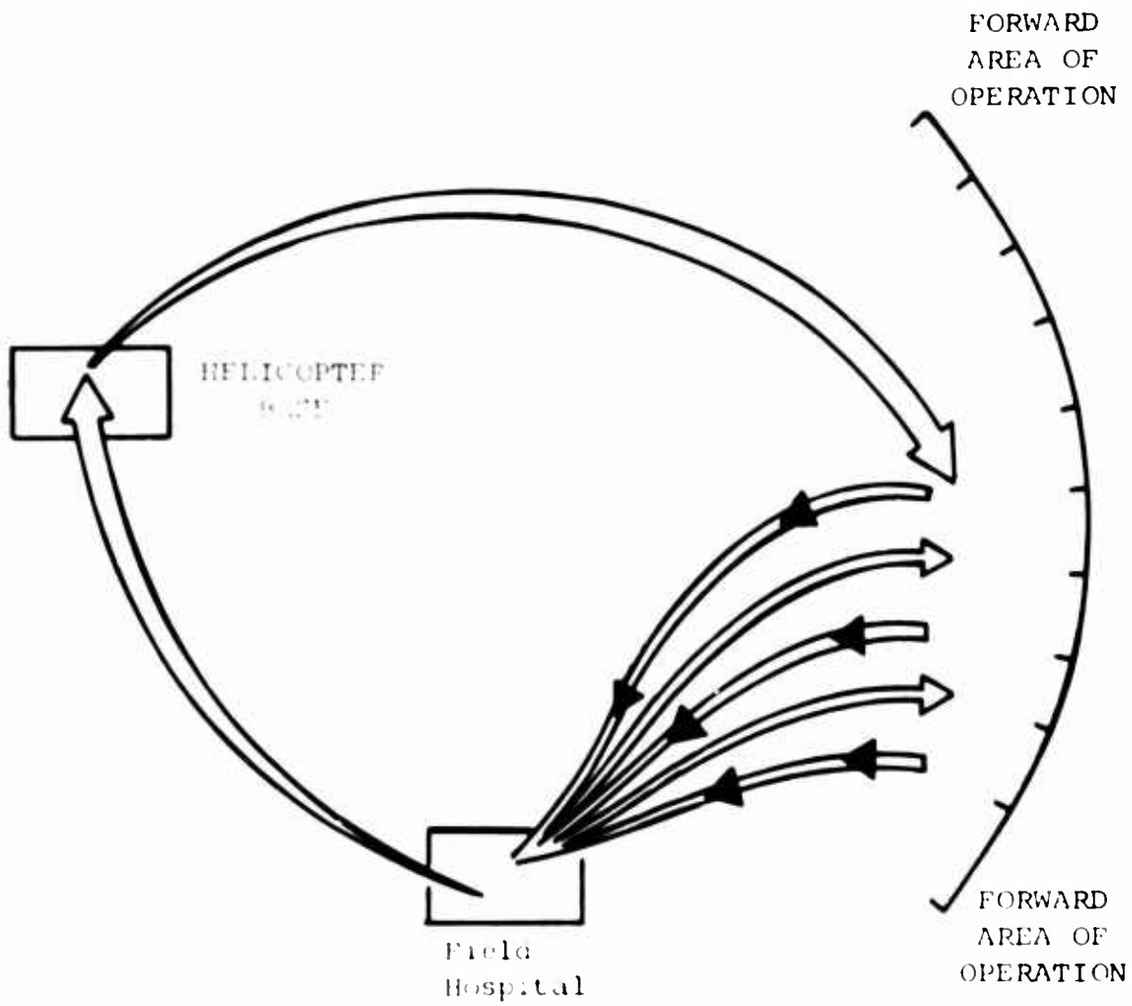


Figure 9. Medical Evacuation Mission.

The helicopter immediately takes off from the helicopter base with one medic aboard and flies to the forward area, where it lands and picks up four litter patients. From here, the helicopter flies to the field hospital, where it lands and discharges the litters. After two more trips to accomplish the evacuation, the helicopter returns from the field hospital to the helicopter base.

The mission demands a payload of 4 litters and 1 medic, and fuel for 150 nautical miles, with 7 takeoffs and landings. Figure 10 presents the cumulative frequency of occurrence of the distance of movement for the medical evacuation mission.

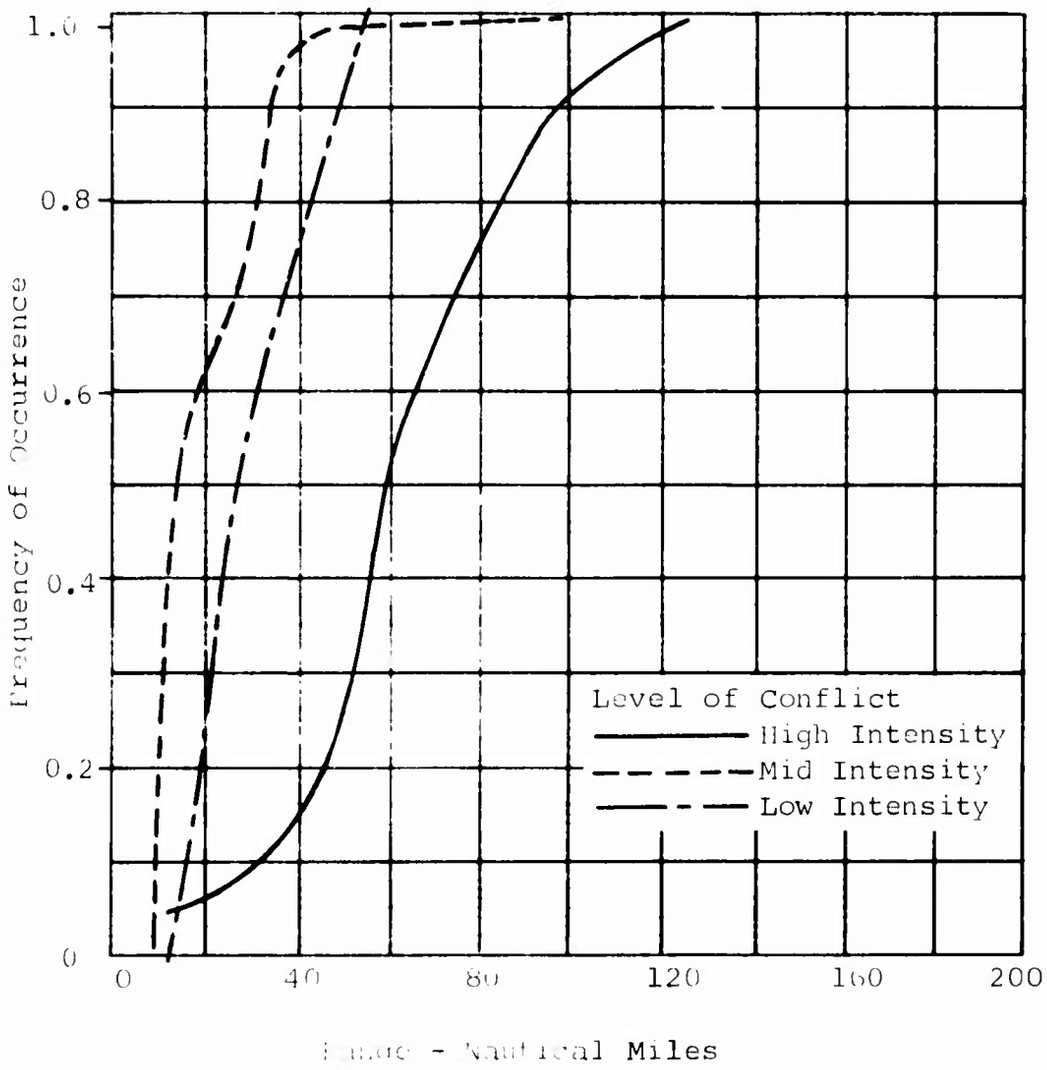


Figure 10. [illegible] Frequency of Occurrence of Distance of Missile [illegible] Nautical Miles [illegible].

The typical mission profile includes:

1. Warm up 2 minutes at Normal Rated Power.
2. Take off and hover out of ground effect, 1/2 minute.
3. Cruise outbound.
4. Land and load 4 litter patients.
5. Take off and hover out of ground effect, 1/2 minute.
6. Cruise inbound.
7. Land and unload litter patients.
8. Repeat 2 through 7 twice.
9. Take off and hover out of ground effect, 1/2 minute.
10. Cruise inbound.
11. Land with 10 percent of initial fuel.

In the typical scenario for the observation mission (Figure 11), the division artillery commander orders continuous daylight observation of the forward area to locate targets and adjust the fire of a general support artillery battalion. Three helicopters are available for this mission. Instructions provide for a continuous observation from dawn to dusk and require that the new relief helicopter be on station prior to the departure of the relieved helicopter. The distance from the helicopter base to the observation area is 10 nautical miles.

The mission demands a payload of one observer, fuel for 20 nautical miles, and endurance for 2 hours. Cumulative frequency of occurrence of flight endurance for the observation mission is plotted in Figure 12.

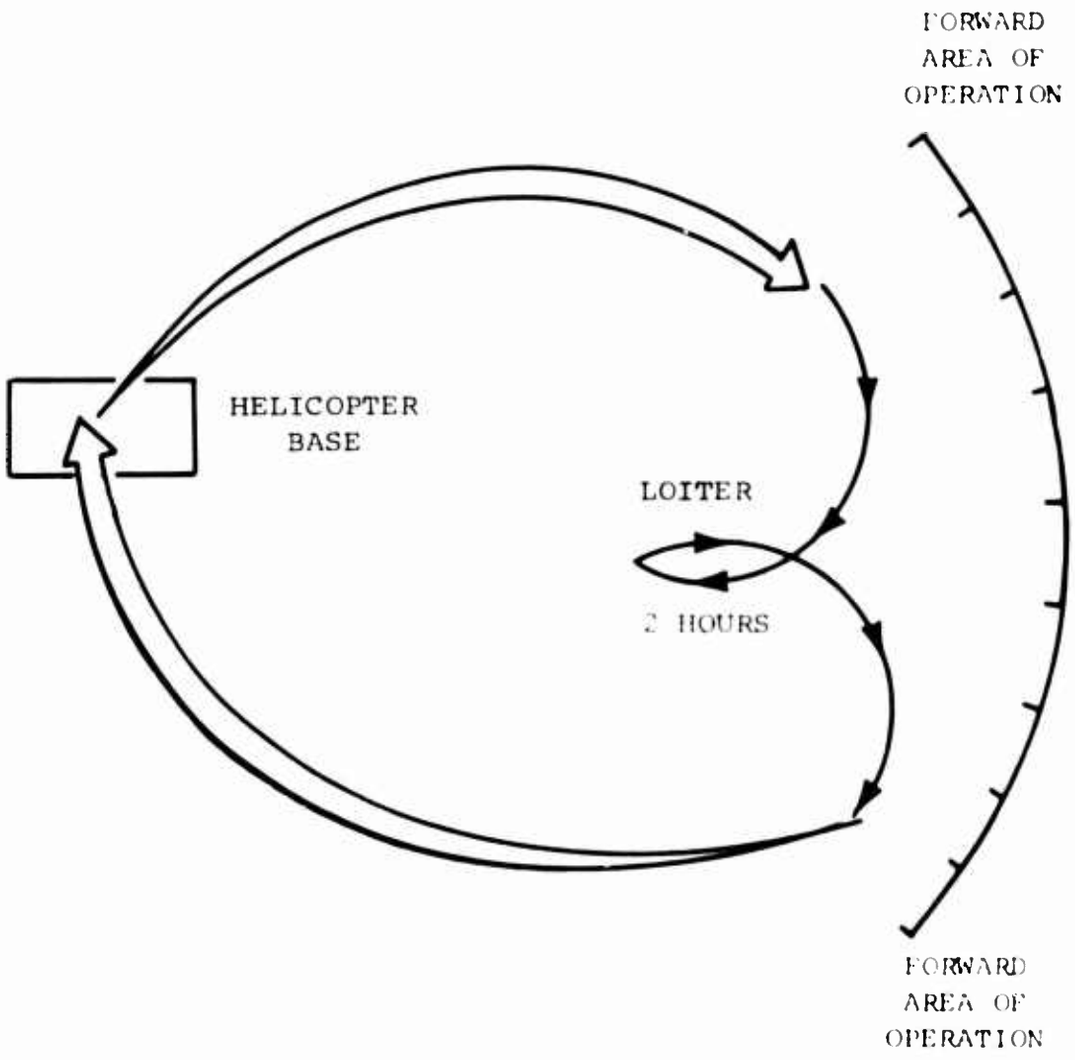


Figure 11. Observation Mission.

The typical mission profile follows:

1. Warm up 2 minutes at Normal Rated Power.
2. Take off and hover out of ground effect, 1/2 minute.
3. Cruise outbound 10 nautical miles.
4. Loiter at best endurance speed for 2 hours.
5. Cruise inbound 10 nautical miles.
6. Land with 10 percent of initial fuel.

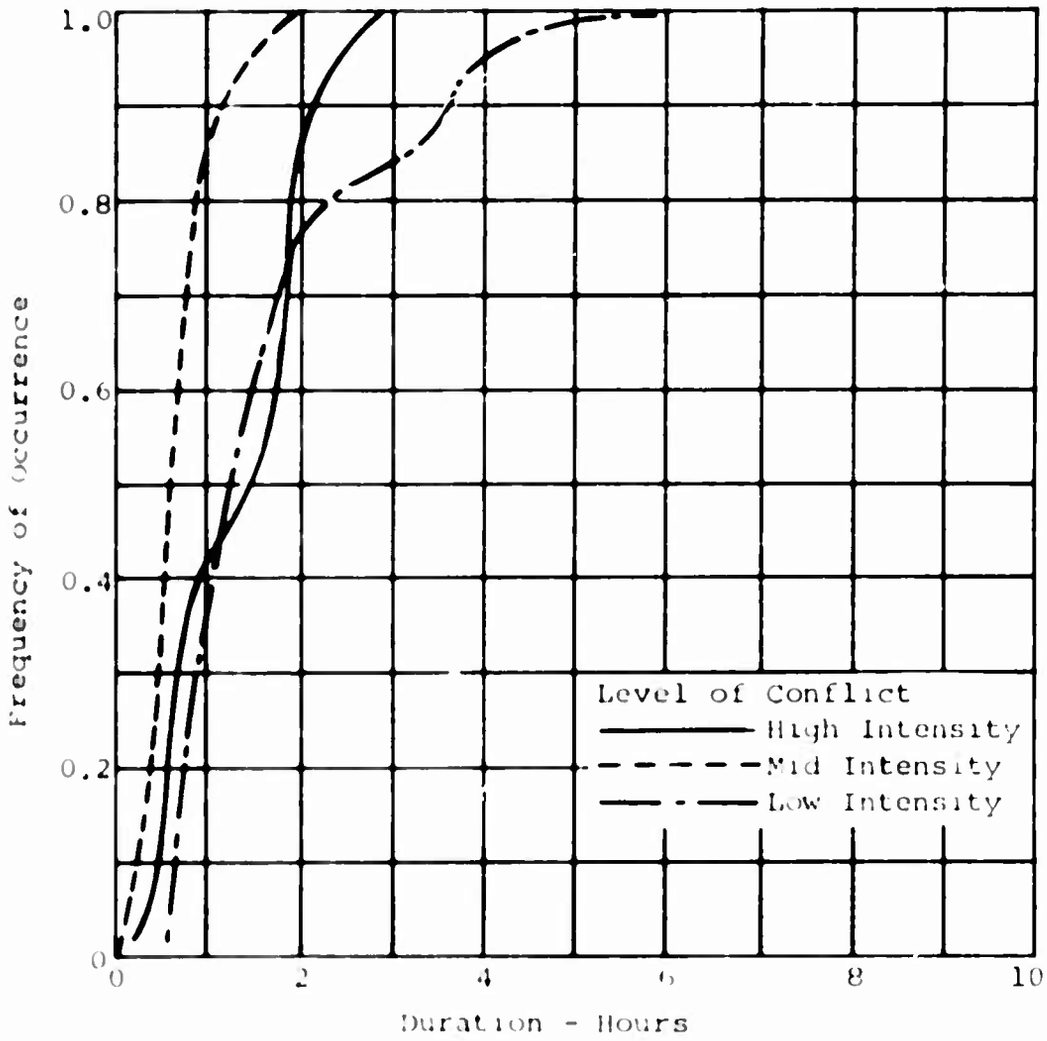


Figure 12. Cumulative Frequency of Occurrence of Flight Endurance for Observation Mission.

TWIN-ENGINE AIRCRAFT

Twin-engine aircraft were designed for the same mission and payload. Definition of one engine inoperative (OEI) capability for these aircraft was based upon evaluation of Army requirements for multiengine aircraft, which established that the helicopter must have go-home capability with one engine inoperative. For the OEI requirements, at 4000 feet, 95°F ambient conditions with the remaining engine at Military Power setting, the aircraft should have the following capabilities

at design gross weight (DGW):

1. Positive rate of climb at OEI cruise speed of 80 knots true airspeed (TAS).

2. Cruise speed, 100 knots TAS in straight, level flight.

3. Landing, after a normal approach and without hover in ground effect.

The service ceiling at 95°F should be 5000 feet or more. Twin-engine aircraft, designed to these OEI requirements, were subsequently evaluated and compared with the single-engine configurations. Aside from the obvious benefits of increased mission reliability, all other factors favored the single engine compared with two engines. This study program was concerned primarily with comparisons between regenerative and nonregenerative engines in an aircraft installation, and such comparisons would be equally valid for single-engine helicopters as for twin-engine ships. To reduce the complexity of the effort, therefore, only single-engine aircraft were retained for consideration in tasks directed toward the final tradeoff studies of weight, performance, reliability, maintainability, and cost parameters. Twin-engine aircraft conceptual designs, and the corresponding data for simple-cycle engines and regenerative engines, have been placed in Appendix I.

TURBOSHAFT ENGINE DESIGN PARAMETERS

Conceptual designs of the utility transport helicopter aircraft in this program initially were based on the data for regenerative and nonregenerative engines of approximately 1000 shp, presented by AiResearch in Reference 1. The design-point component efficiencies, losses, and cooling-air flows for these engines were somewhat optimistic for an engine that could be available in 1975. Basically, the design parameters used in Reference 1 were for an airflow of 5.0 lb/sec, and they resulted in a specific power (shaft horsepower divided by compressor inlet airflow, shp/lb/sec) and a specific fuel consumption (SFC) that represented extremely difficult targets for the time frame considered. This optimism was also reflected in the very high power-to-weight ratio of the non-regenerative simple-cycle engine. The advanced component technologies would require substantial development effort to achieve the desired production status. Therefore, to provide a standard of comparison for the advanced-technology simple-cycle engine, Boeing redefined, by agreement with the Army, performance and weight characteristics typical of turboshaft engines under development or proposed for development utilizing available technologies. Component trend data were developed and used to define design-point component performance and to generate overall performance for a simple-cycle engine consistent with correlations of specific power and SFC for available technology engines.

The output shaft was located at the rear of both the regenerative and nonregenerative engine configurations in Reference 1, providing a simplified shafting and bearing system. This arrangement was adopted as the standard for the study and had the added advantage of minimizing shaft dynamics problems. In addition, by isolating each rotating assembly, turbine tip clearances could be reduced to maximize turbine efficiencies.

The power requirement of the utility helicopter considered in this study was approximately 1000 shp, essentially the size of the engines described in Reference 1. The twin-engine aircraft sized for the same mission and payload, however, dictated an engine power approximately one-half of this 1000-shp capability. The characteristics of the smaller advanced-technology engine - approximately 500 shp - were obtained by downgrading the performance and weight data for the 1000-shp engine, while the characteristics of the 500-shp available-technology engine were developed from the applicable

trend curves. Although not used in the final studies, data for the 500-shp engines and the twin-engine aircraft have been assembled in Appendix I for reference purposes. This section of the report considers only the characteristics of simple-cycle turboshaft engines of approximately 1000 shp, utilizing both advanced technologies and available technologies.

ADVANCED-TECHNOLOGY ENGINE

Engines used in the nonregenerative portion of the study were defined in Table III. In the first column, component design-point data and specific power and SFC for the advanced-technology turboshaft engine were reproduced from Reference 1. It should be noted that although the 9:1 compressor pressure ratio is optimum for the regenerative engine, the simple-cycle engine optimizes at a higher pressure ratio. To determine the optimum compressor pressure ratio for the advanced-technology simple-cycle engine, trend data were developed for component design-point performance parameters and used in a parametric study of the effect of pressure ratio on engine performance. A design-point adiabatic efficiency trend, typical of advanced-technology compressors in the 5.0-lb/sec class, was plotted in Figure 13. The efficiency value of 0.82 at pressure ratio 9:1, from Reference 1, was on this curve, and the slope of the curve corresponded very nearly to a constant polytropic efficiency.

In Figure 14, an approximate parametric trend was developed for two-stage turbines in the same size class. Using the power turbine efficiency and pressure ratio from Reference 1 (Table III, first column) and a slope parallel to the typical efficiency curve, trend data were developed for advanced-technology two-stage turbines. The engine designs in Reference 1 used a single-stage gas-generator turbine having relatively high loading. The efficiency of this stage was corrected for the effect of aerodynamic losses (due to cooling), and the assumed efficiency for the developed stage was reduced to 0.86, the point plotted in Figure 14. The efficiencies of the two-stage gas-generator turbines considered in the parametric study also were downgraded slightly from the advanced-technology curve to account for cooling air. The trend curves for cooling requirements and leakage flows were plotted in Figure 15. Leakage flows were proportional to pressure ratio, while the cooling air requirements at constant turbine-inlet temperature were dependent on compressor exit temperature, which also is a function of pressure ratio.

TABLE III. SIMPLE-CYCLE TURBOSHAFT ENGINE DESIGN-POINT PARAMETERS

Parameter	Advanced Technology (Ref. 1)	Advanced Technology, Optimum Pressure Ratio	Available Technology
Compressor			
Inlet Airflow, lb/sec	5.0	5.0	5.10
Pressure Ratio	9.0	14.0	14.0
Adiabatic Efficiency*	.82	.808	.79
Exit Temperature, °F	601.	761.	776.
Cooling Air/Inlet Airflow	.035	.035	.057
Leakage/Inlet Airflow*	.015	.031	.023
Combustor			
Efficiency*	.99	.99	.985
Fuel/Compressor Inlet Airflow	.0265	.0238	.0217
Pressure Loss*	.03	.03	.05
Gas Generator Turbine			
Inlet Temperature, °F	2300.	2300.	2200.
Inlet Flow, lb/sec	4.883	4.789	5.744
Mechanical Efficiency*	.975	.975	.995
Exit Temperature, °F	1831.	1673.	1556.
Adiabatic Efficiency*	.86	.88	.86
Pressure Ratio	2.58	3.61	4.09
Interstage Turbine Diffuser			
Pressure Loss*	.02	.02	-
Temperature, °F (Cooling-Air Mixed)	1815.	1662.	1512.
Power Turbine			
Inlet Temperature, °F	1815.	1662.	1512.
Inlet Flow, lb/sec	5.033	4.964	6.091
Exit Temperature, °F	1315.	1158.	1096.
Adiabatic Efficiency*	.90	.89	.875
Pressure Ratio	3.22	3.59	3.07
Exhaust Diffuser			
Pressure Ratio	1.03	1.03	1.058
Specific Power, hp/lb/sec	204.0	200.0	164.0
Shaft Power, hp	1020.	1000.	1000.
SFC, lb/hr/hp	.468	.429	.475

*Efficiencies, pressure losses, bleed and leakage flows expressed as decimal fractions.

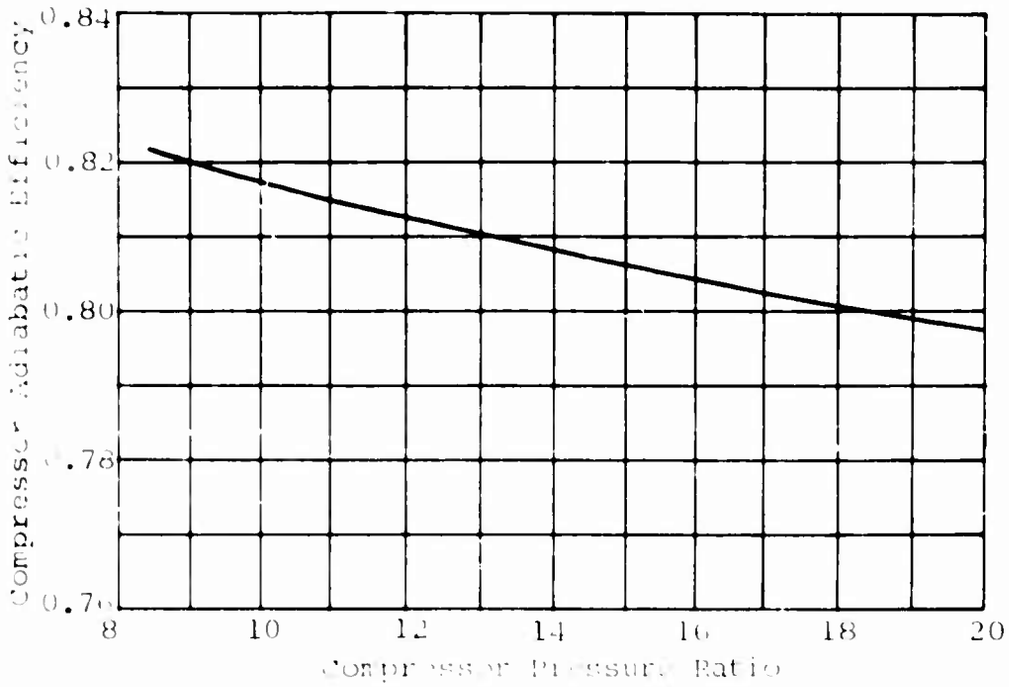


Figure 13. Compressor Design-Point Performance for 1000-SHP Advanced-Technology Engine.

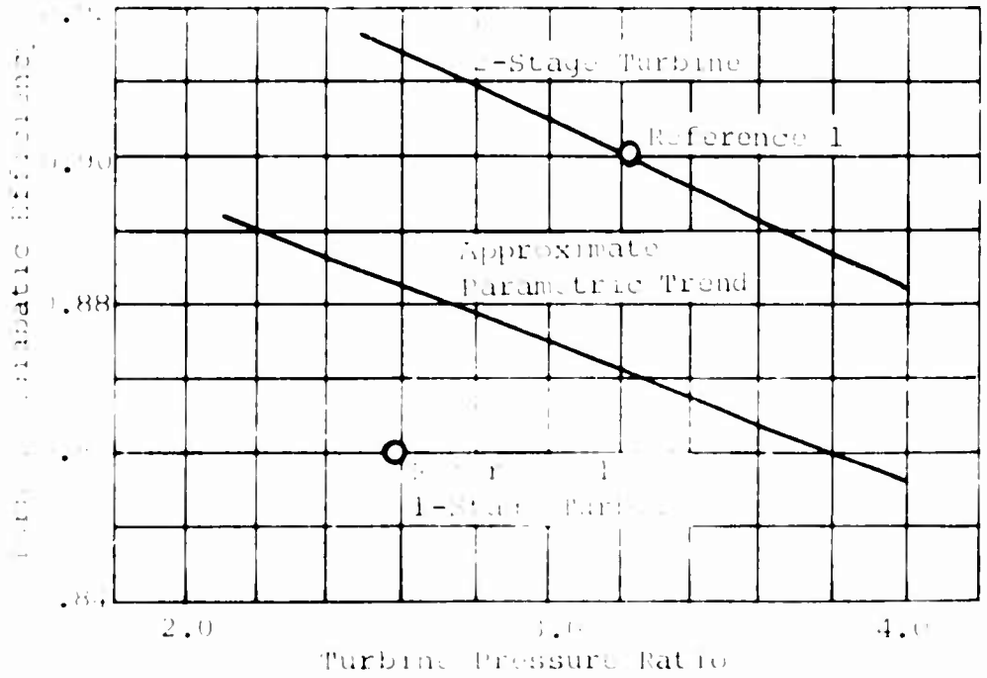


Figure 14. Turbine Design-Point Performance for 1000-SHP Advanced-Technology Engine.

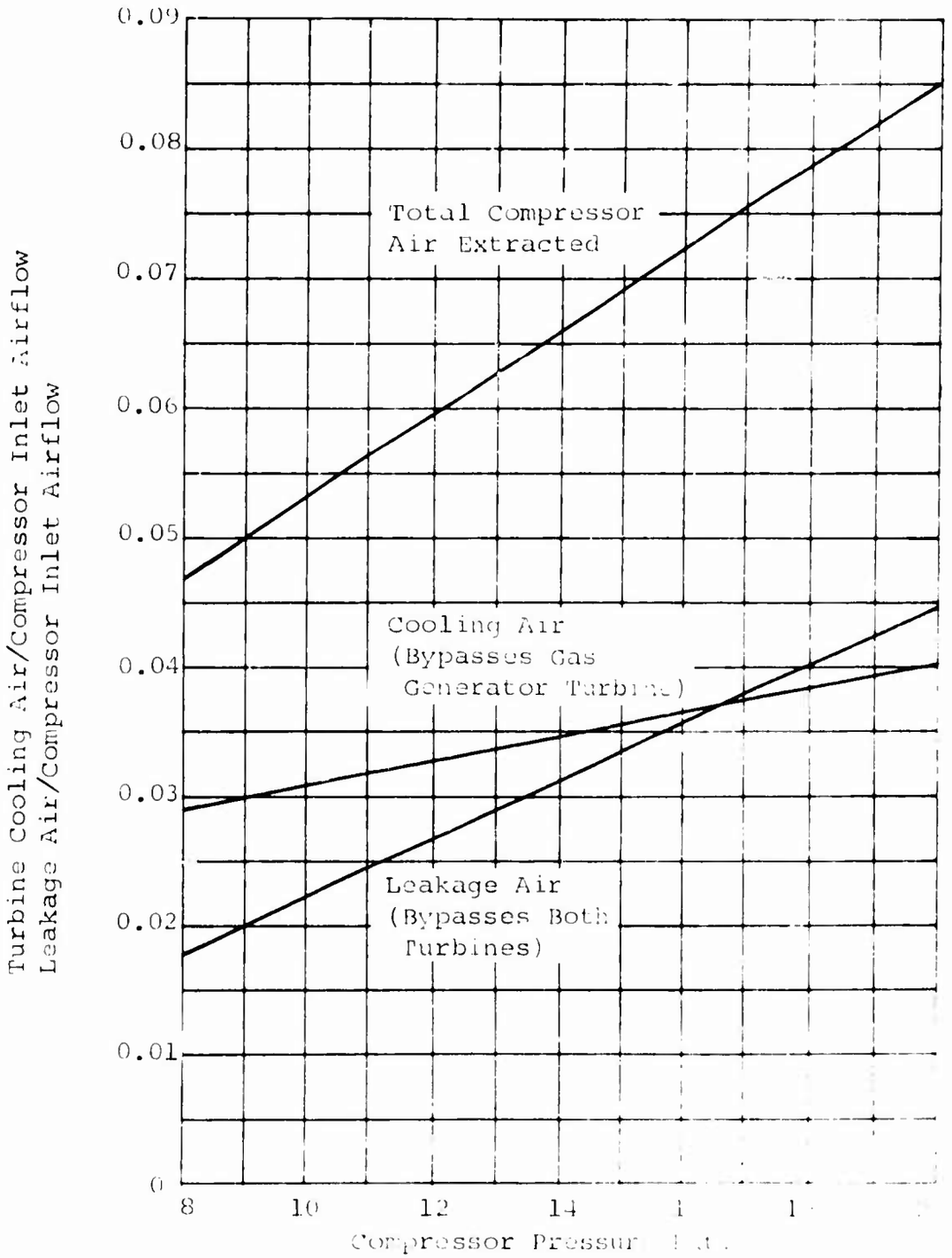


Figure 15. Turbine Cooling Air and Leakage Flow for 1000-SHP Advanced-Technology Engine.

The trends for compressor and turbine efficiencies and for cooling air and leakage flows, in Figures 13,14, and 15, were used in the parametric study; Figure 16 illustrates the resulting specific power and SFC. It is recognized that higher values of specific power reflect smaller physical size of the engine for a fixed power requirement. In most applications, however, the impact of fuel load on the aircraft gross weight is significantly greater than that of engine weight. For this reason, greatest emphasis was placed on optimizing the SFC of the simple-cycle engine, and a pressure ratio of 14:1 was selected for the design point using advanced-technology components.

Design-Point Performance

The second column in Table III presents component data and engine performance for the design point of the advanced-technology simple-cycle engine. Compressor performance, cooling air, and leakage parameters were taken from Figures 13 and 15, and combustor performance was assumed to be the same as the Reference 1 data. The pressure ratio across the gas-generator turbine (3.61:1) dictated a two-stage design, and the efficiency of the gas-generator turbine was degraded somewhat from the trend data in Figure 14 to account for the impact of turbine cooling flows on performance. Other efficiencies and losses were assumed to be the same as the data from Reference 1 or were read from the applicable trend curves.

The SFC was substantially better than that of the non-recuperative engine design presented in Reference 1 (0.429 lb/hr/hp compared to 0.466 lb/hr/hp).

Off-Design Performance

Off-design performance was developed for the simple-cycle engine, to be used in performance calculations for the utility transport aircraft over the complete spectrum of altitude-ambient temperature conditions, forward flight speeds, and power requirements. The compressor performance map assumed for this calculation is shown in Figure 17. Typical two-stage turbine performance characteristics were used to generate the power turbine efficiency trend at optimum shaft speed, plotted in Figure 18. Figure 19 presents the results of the off-design performance calculation. Referred shaft horsepower and referred fuel flow are shown as functions of referred turbine-inlet temperature for various forward flight speeds - at

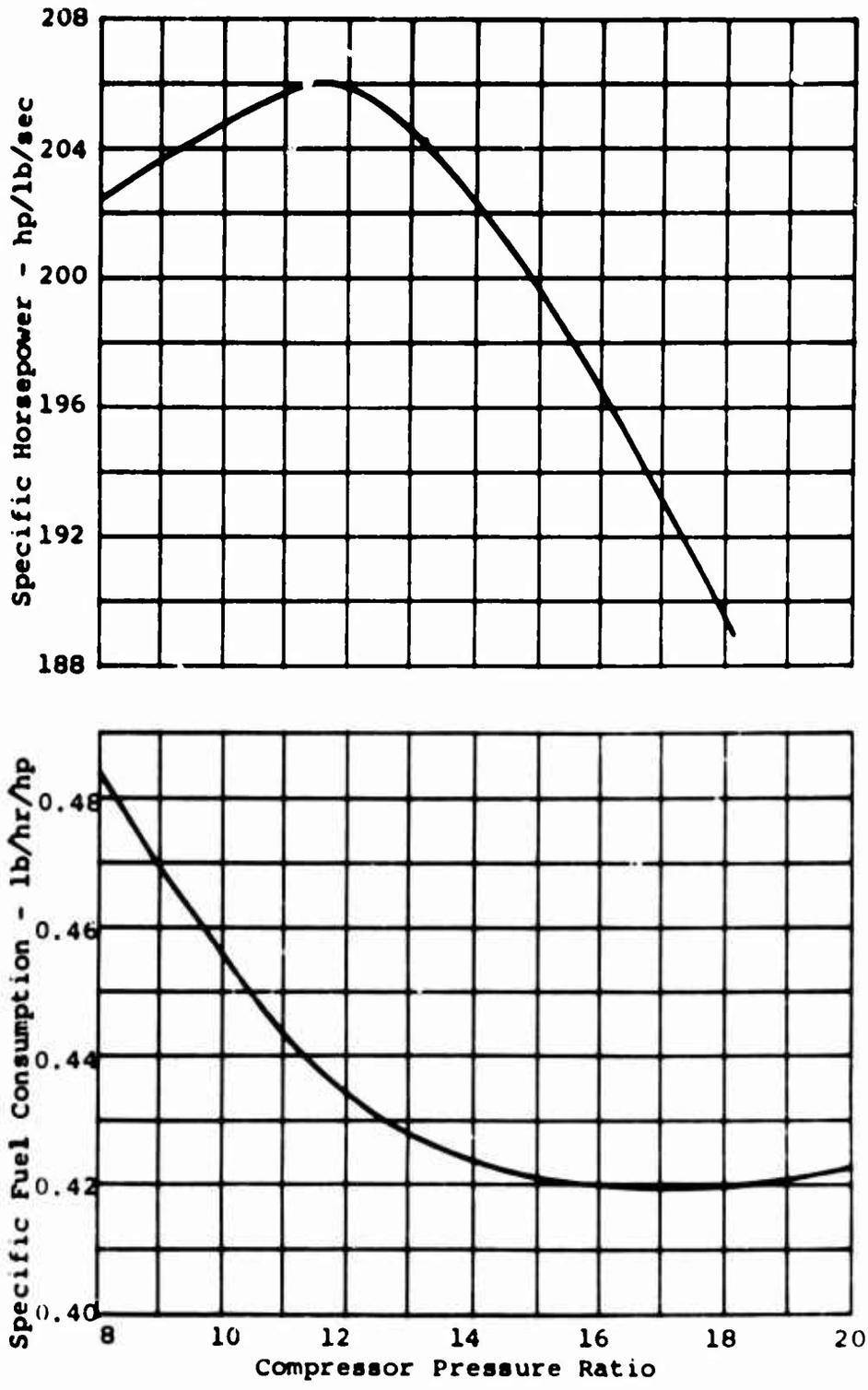


Figure 16. Design-Point Performance of Parametric Advanced-Technology Engines (TIT = 2300°F).

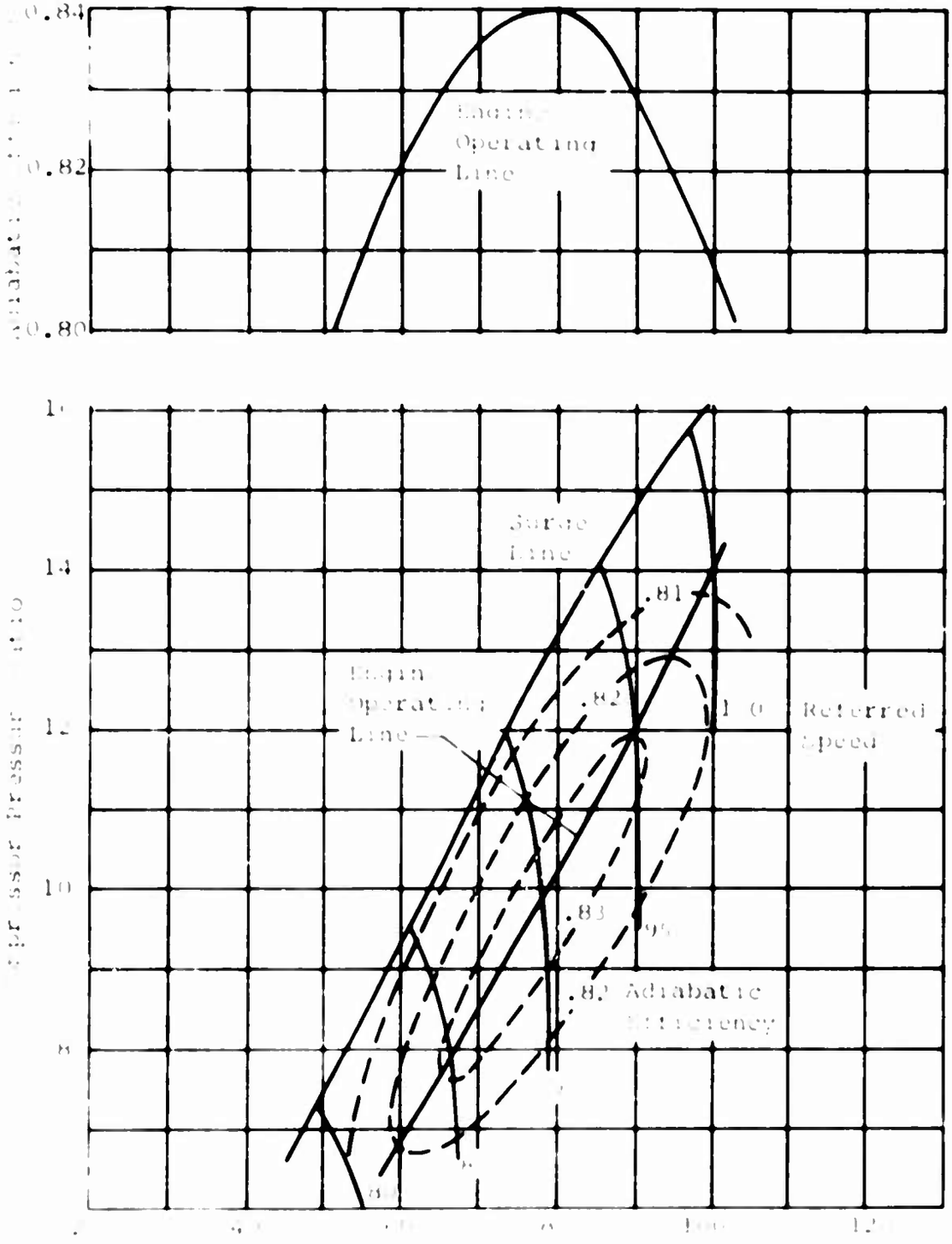

0.84
0.82
0.80
Surge
Line
.81
.82
1.0 Referred
Speed
.83
.85
.8? Adiabatic
Efficiency
14
12
10
8

Figure 17. [illegible]

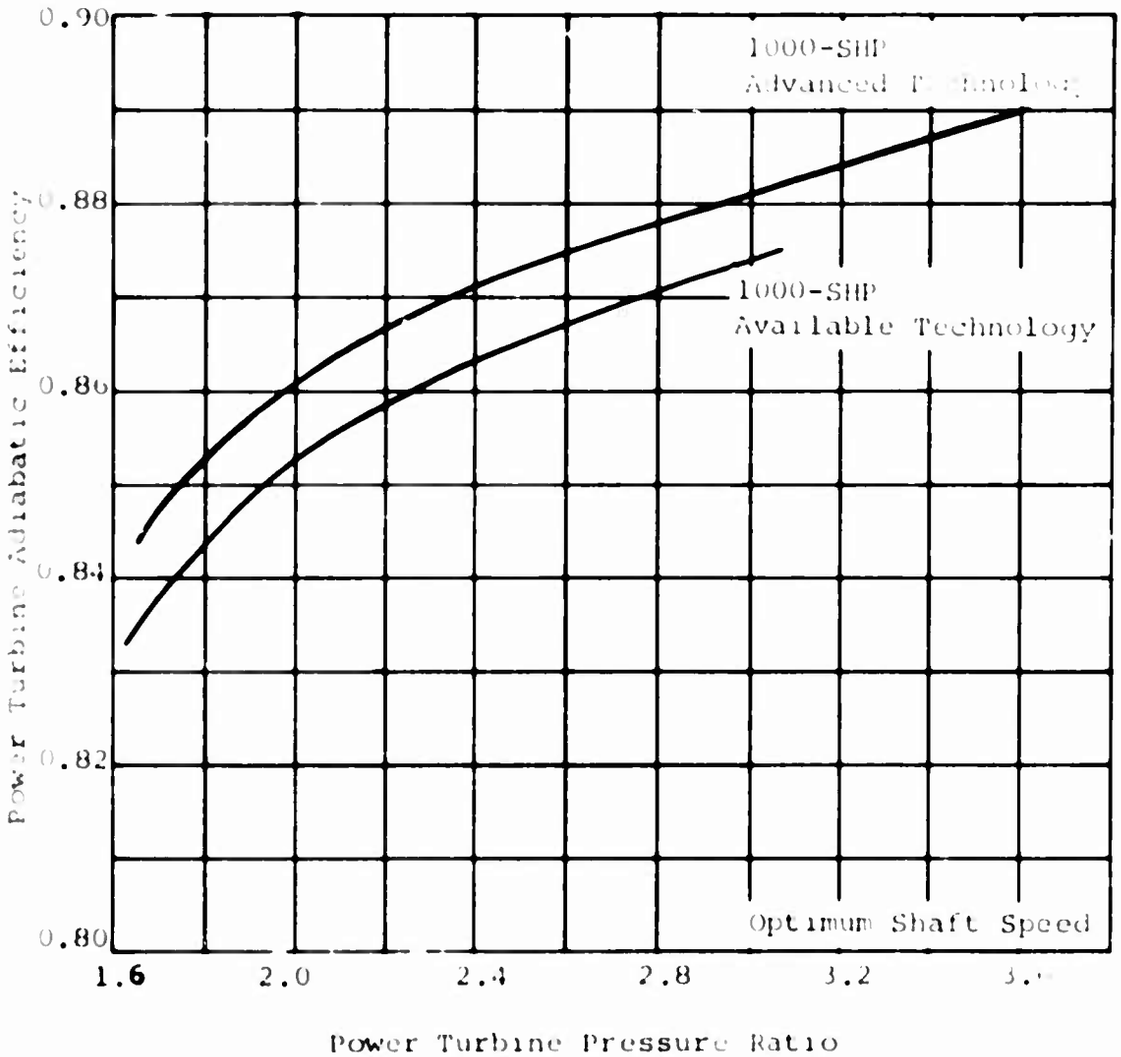


Figure 18. Power Turbine Off-Design Efficiency Trend.

optimum output shaft speed. Typical trends of optimum output shaft speed as a function of power are given in Figure 20, and power corrections for nonoptimum shaft speed are shown in Figure 21. The data in Figure 20 were normalized and are presented as a percentage of design-point values, to [illegible] as typical trend data for the characteristics of all the non-regenerative and regenerative engines. Figures 20 and 21 were used to calculate power and SFC for constant helicopter rotor speed operation, using as a basis the optimum shaft speed performance in Figure 19.

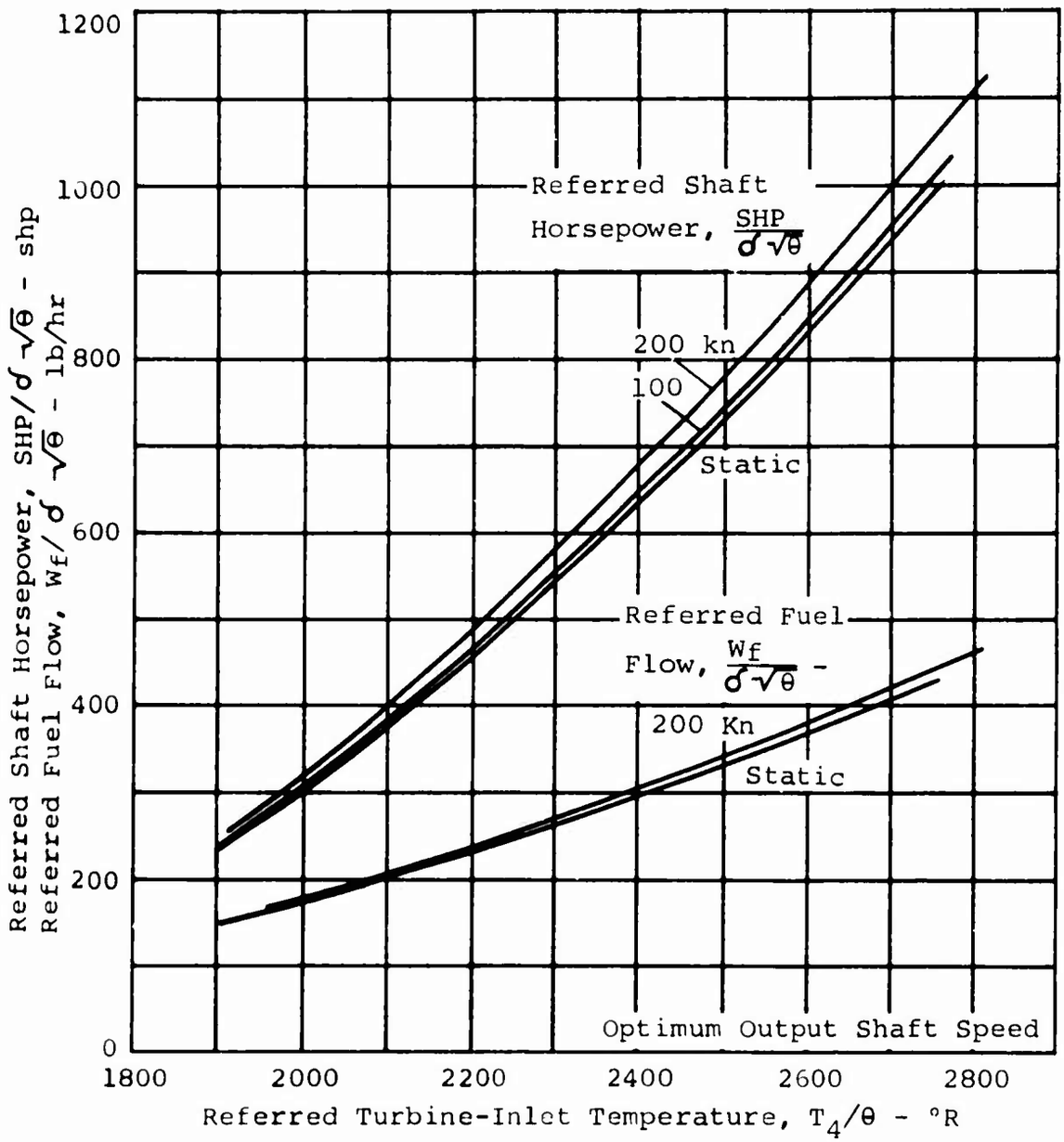


Figure 19. 1000-SHP Advanced-Technology Simple-Cycle Engine Performance.

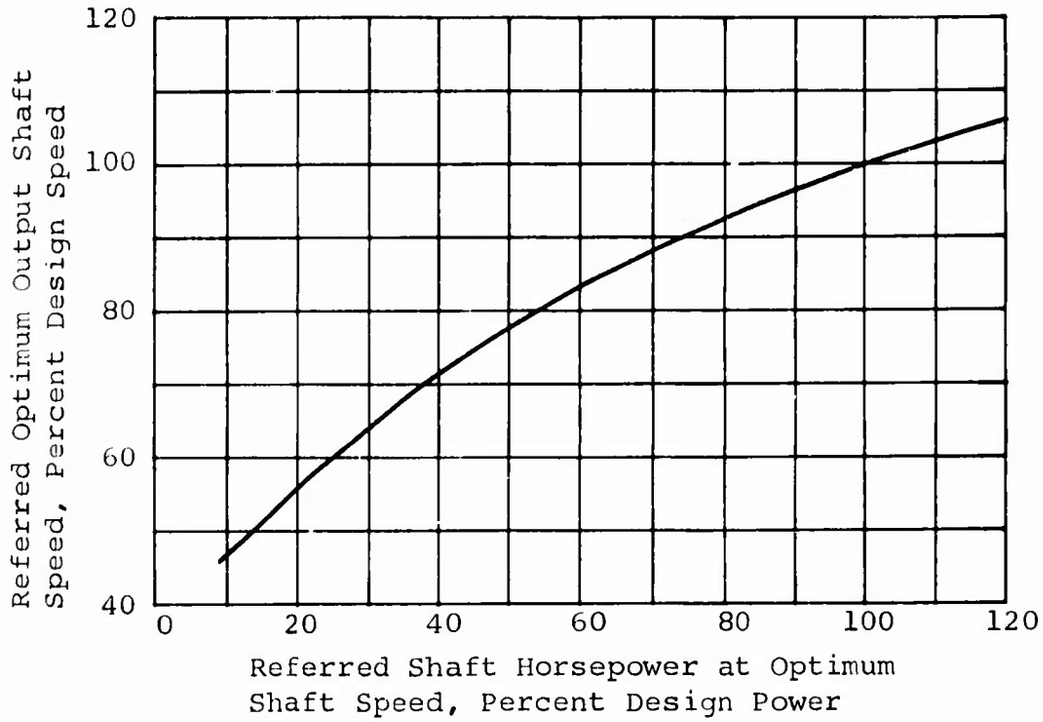


Figure 20. Optimum Output Shaft Speed as a Function of Shaft Horsepower.

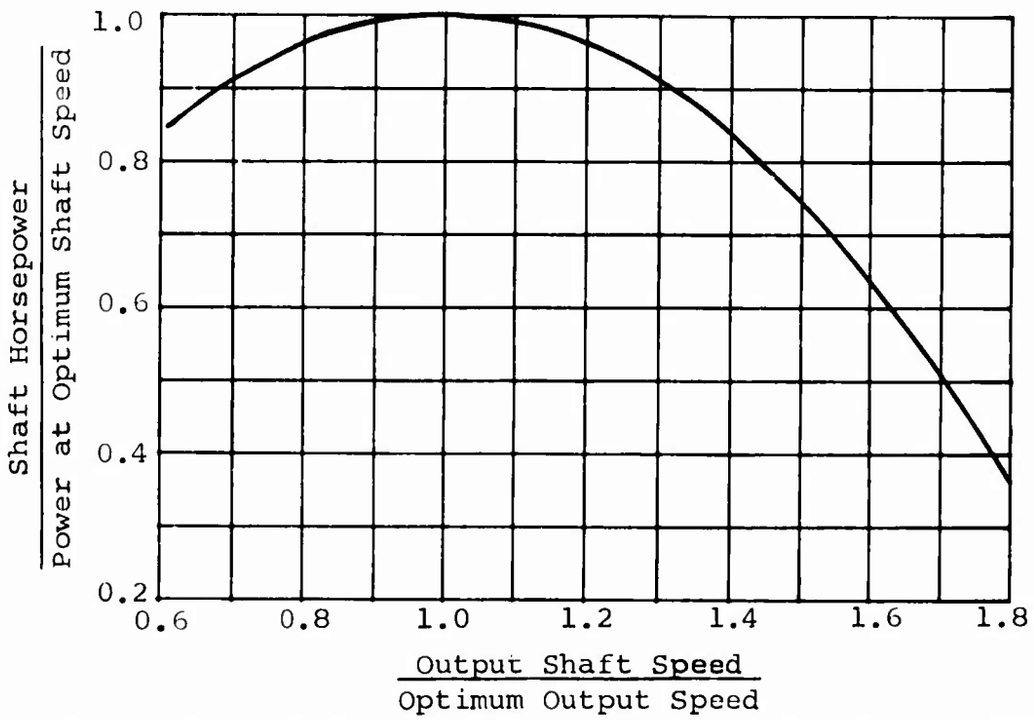


Figure 21. Shaft Horsepower Correction for Nonoptimum Shaft Speed.

Ratings for the advanced-technology engine were predicated upon the rating philosophy of a number of engines presently under development, in which the shaft horsepower at the NRP settings was 85 to 87 percent of the power at the MRP setting. This corresponded to approximately a 150°F difference in turbine-inlet temperature between the two ratings. Accordingly, the turbine-inlet temperature of the advanced-technology simple-cycle engine was selected as 2150°F at NRP and 2300°F at MRP.

Engine Configuration

The engine configurations developed in Reference 1 had compressors with one axial and one radial stage, a single-stage gas generator turbine, and a two-stage power turbine. For the higher pressure ratio (14:1) selected for the advanced-technology simple-cycle engine, a compressor with two axial stages and one radial stage and a two-stage gas-generator turbine was judged to be more suitable. The arrangement of components for this engine is pictured in Figure 22.

Because only rear-drive engines were used in the study, the simple-cycle engine would have a curved exhaust duct similar to the General Electric T58 engine. For convenience, therefore, the T58 exhaust duct dimensions were scaled to fit the engines.

Engine Weight

The dry weight for the nonrecuperative engine in Reference 1 was 110 pounds for a 962-shp engine. Since the range of powers considered in scaling the engine to 1000 shp was quite narrow, linear scaling in proportion to the power was used to calculate a dry weight of 114 pounds for a 1000-shp engine. The compressor and gas-generator turbine configuration changes mentioned above resulted in a further increase in weight (approximately 17%), and 129 pounds was selected as the dry weight of the advanced-technology simple-cycle engine.

Engine Reliability and Maintainability

Reliability and maintainability data for the engines have been included with data for the other aircraft subsystems in the Aircraft Comparative Analyses section of the report.

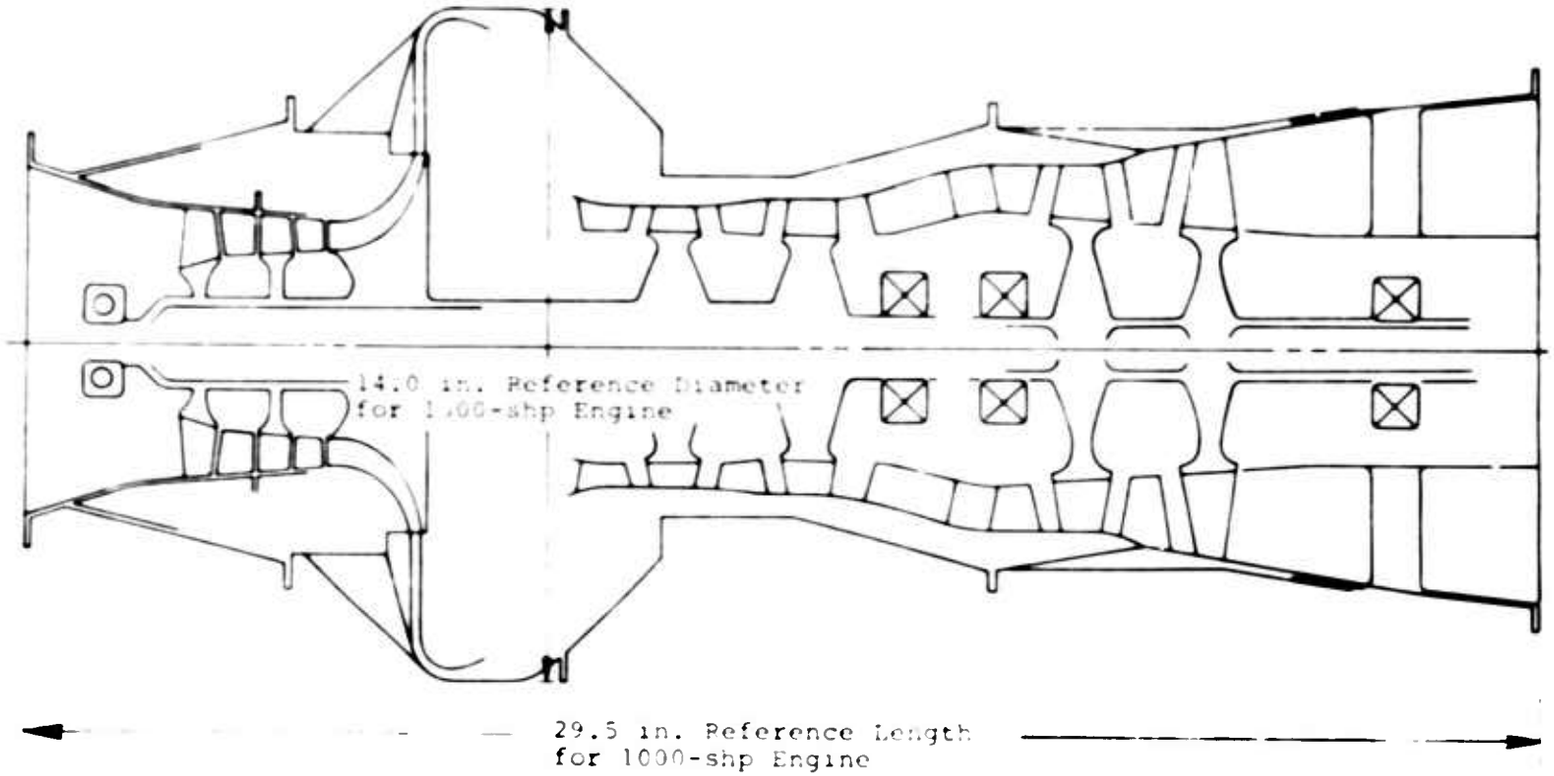


Figure 22. Advanced-Technology Simple-Cycle Engine.

Engine Cost

The pioneering effort in correlating development and procurement cost of aircraft turbine engines was undertaken by the RAND Corporation, and the results were published in Reference 6. Although attempts were made to relate cost to several technical variables, satisfactory relationships were derived only for cost as a function of engine size or shaft horsepower. The RAND document defined development cost to include initial contractor preliminary design, subsequent engineering, prototype tooling, material, fabrication, assembly and bench testing of scale or full-size components or complete engines to and including qualification testing to military acceptance specifications, and also production tooling. Procurement cost was defined as the total cost to fabricate and assemble complete engines, including labor, material, overhead, and profit. Also included was sustaining tooling, factory liaison, acceptance testing, and the preparation of supporting data such as maintenance manuals.

Following this same line of thought, Boeing has added a large amount of data to the original RAND correlation, and the cost curves have been updated to reflect 1970 dollars. Resulting development costs for turboshaft engines are shown in Figure 23. Procurement costs for turboshaft engines are provided in Figure 24, the wide band covering virtually all available data points. The production cost data correlated in Figure 24 were cumulative average costs for 1000 engines in 1970 dollars, based on a 92 percent learning curve and 3 percent average cost increase per year. Unit costs to be used in the Systems Engineering Model were based on the same 92 percent learning curve, with cumulative average costs developed by extrapolating the data in Figure 24. The data point for the nonregenerative engine from Reference 1, which was plotted in Figure 24, proved to be on the low side of the spread of cost data, but the data for the simple-cycle engines considered in this study were developed using this cost trend, as illustrated by the line plotted intersecting the Reference 1 data point.

AVAILABLE-TECHNOLOGY ENGINE

Boeing surveyed other engine manufacturers to determine the state of the art in small turboshaft engines under development or proposed for development, and correlations of component data were generated for these engines, using available technologies. The component trend data were used to define the design-point

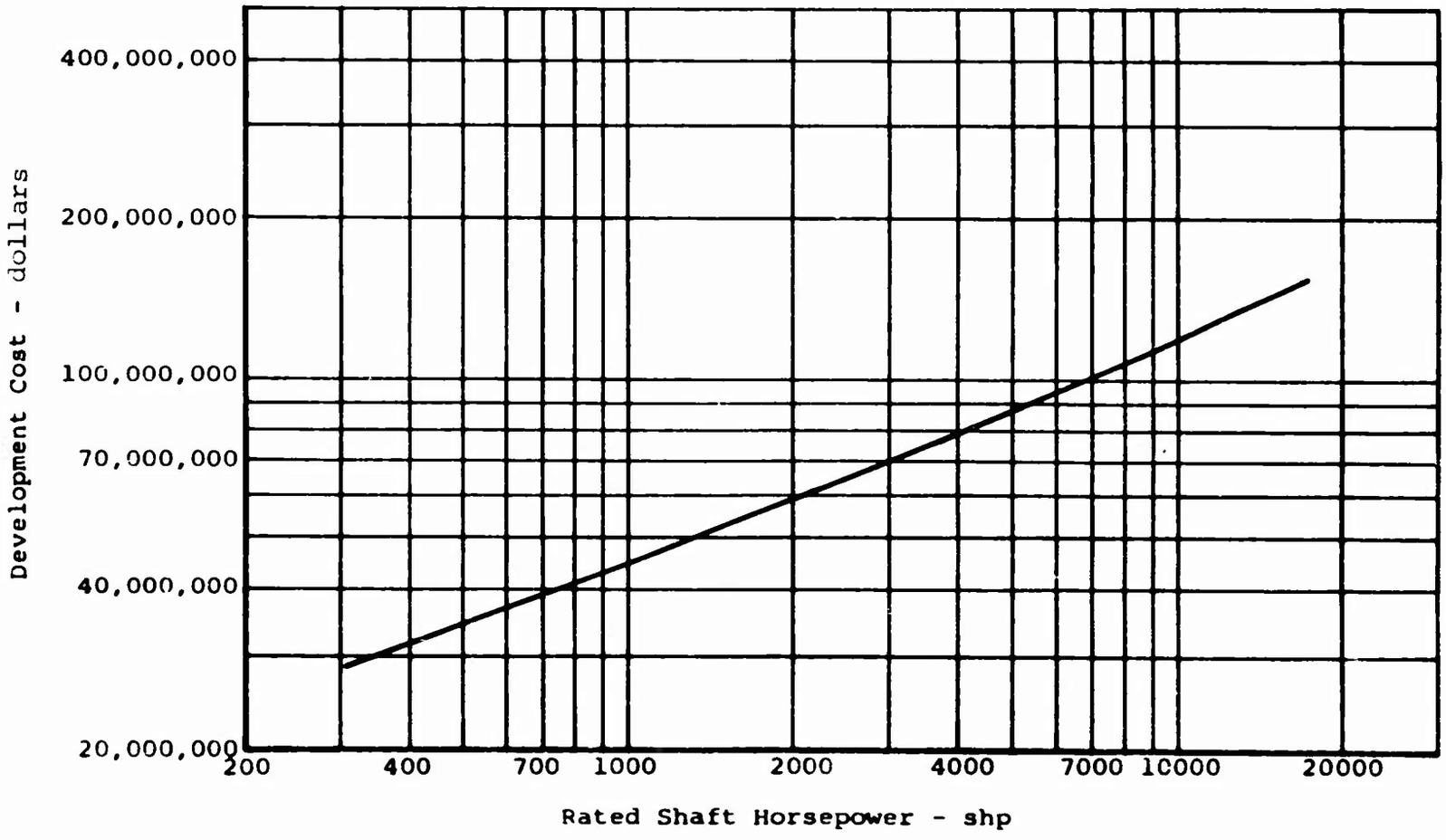


Figure 23. Development Cost (Including Production Tooling) for Simple-Cycle Turboshaft Engines, 1970 Dollars.

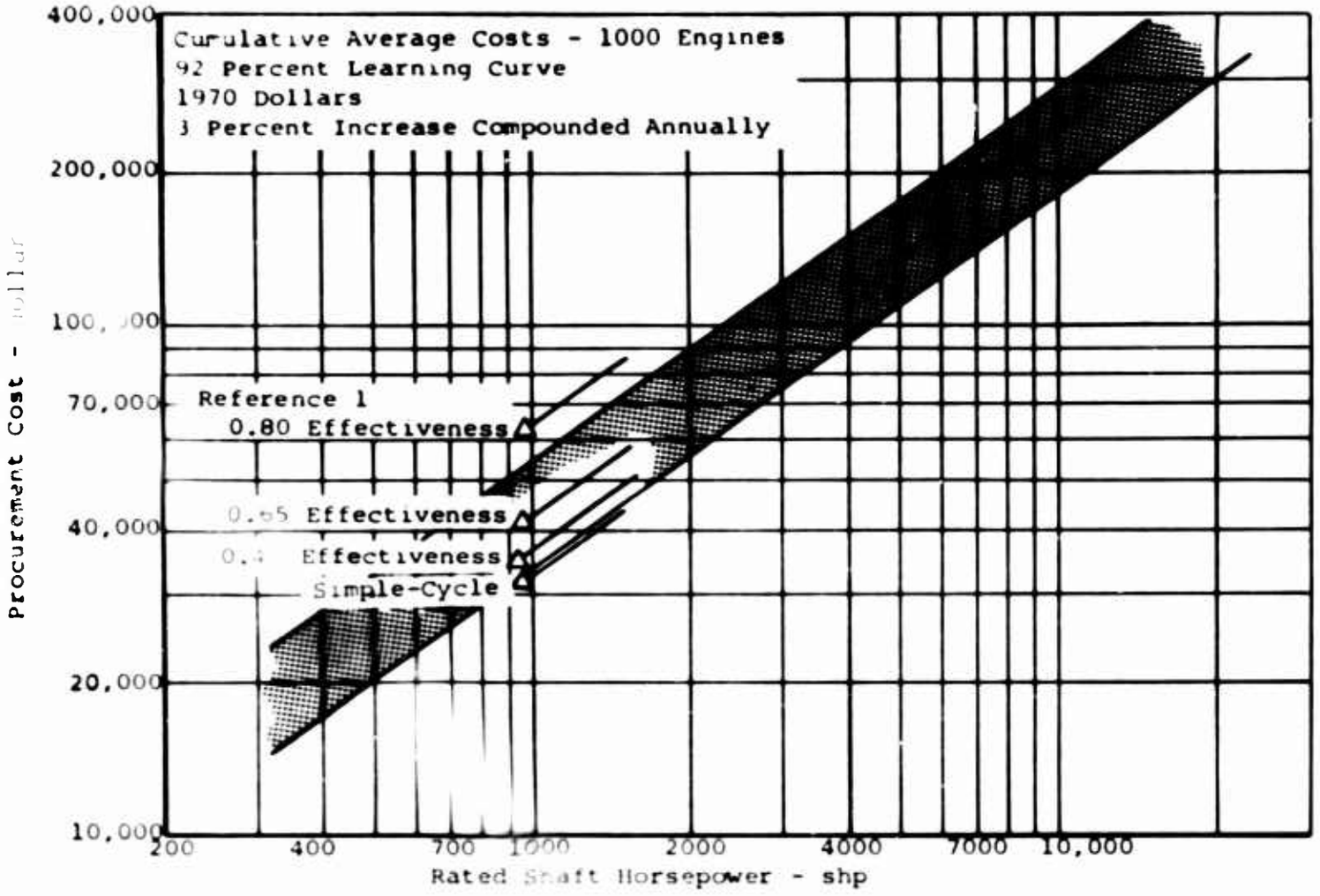


Figure 24. Calculated Average Procurement Cost for Simple-Cycle Turboshaft Engines, With Point Data and Trends for Advanced-Technology Regenerative and Intercooled-Regenerative Engines From Reference 1.

component performance for this simple-cycle engine and to calculate overall performance, which was consistent with trends of specific power and SFC. Performance and weight characteristics were determined for the available-technology turboshaft engine to provide a standard of comparison for the advanced-technology engine.

Design-Point Performance

Compressor efficiency trends were plotted in Figure 25, the available engine data falling generally in the airflow ranges of 3 to 4 lb/sec or 7 to 10 lb/sec. Similarly, turbine efficiency trends for these two size ranges were plotted in Figure 26. The design airflow of the available-technology simple-cycle engine - 9.10 lb/sec required to produce 1000 shp - led to the selection of design-point component efficiencies indicated by a symbol in Figures 25 and 26. Other efficiencies, losses, and cooling-air flows were comparable to data given for small turboshaft engines under development.

The design pressure ratio of 14:1 was consistent with what could be expected for a 1000-shp engine utilizing available technologies. As expected, engines presently under development in the 1500-shp class or larger had higher pressure ratios, while smaller engines in the 3- to 4-lb/sec airflow range had pressure ratios from 8:1 to 10:1. In contrast, the 2300°F turbine-inlet temperature of the AiResearch advanced-technology engine was higher than what could be anticipated for a 1000-shp available-technology engine. A design turbine-inlet temperature of 2200°F was selected as more representative for this engine.

The component performance data and the resulting specific power and SFC for the available-technology engine are presented in the third column of Table III. Specific power and SFC were compared with the trends developed for small turboshaft engines in Figures 27 and 28, to establish the validity of the calculated data. This engine was used in subsequent study tasks as the available-technology simple-cycle engine.

Off-Design Performance

The compressor performance map used for the off-design performance calculations of the available-technology turboshaft engine was the same as the one used previously (Figure 17), but the efficiency throughout was downgraded by 0.5% to reflect

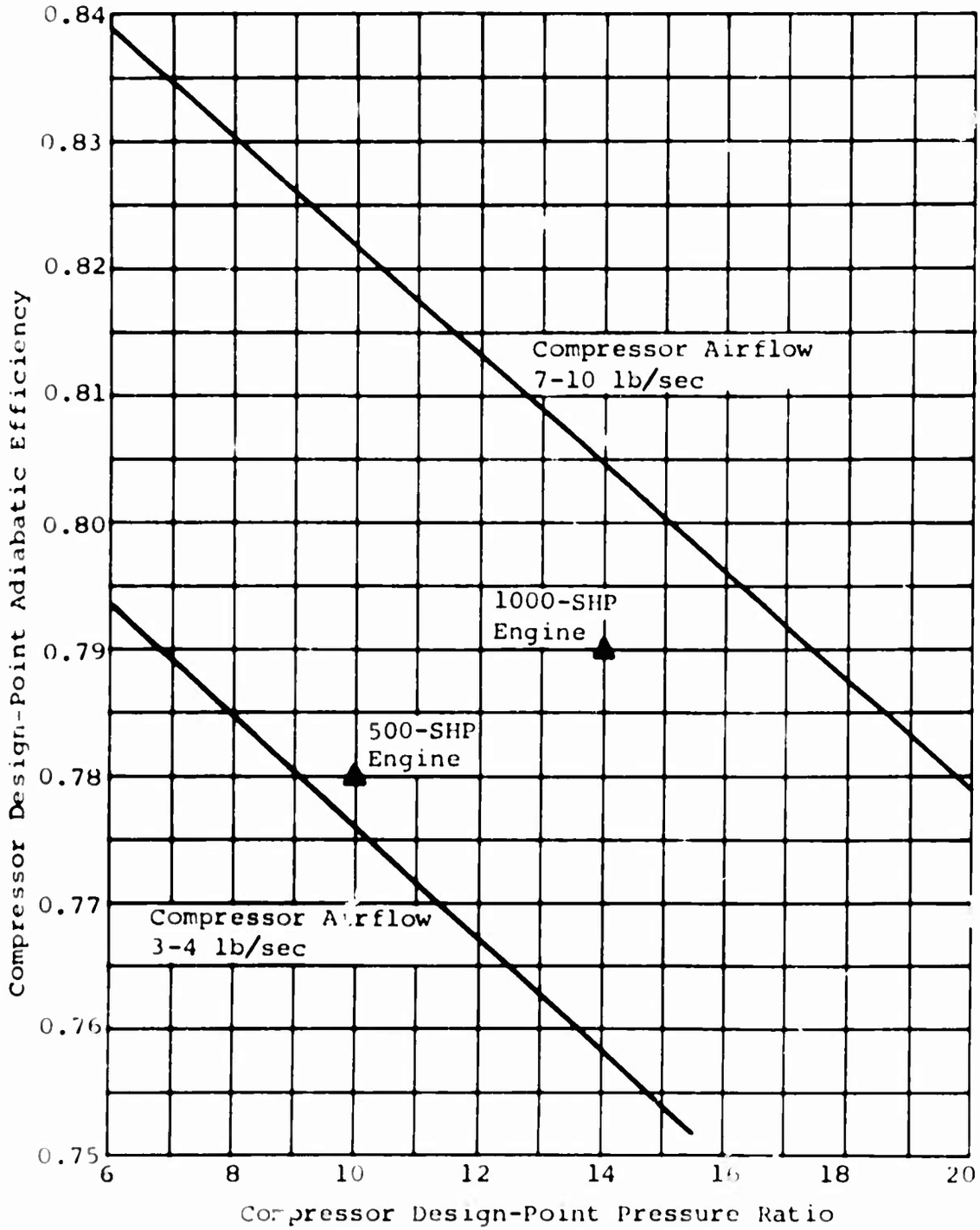


Figure 25. Compressor Design-Point Efficiency Trends for Small Turboshaft Engines Utilizing Available Technologies.

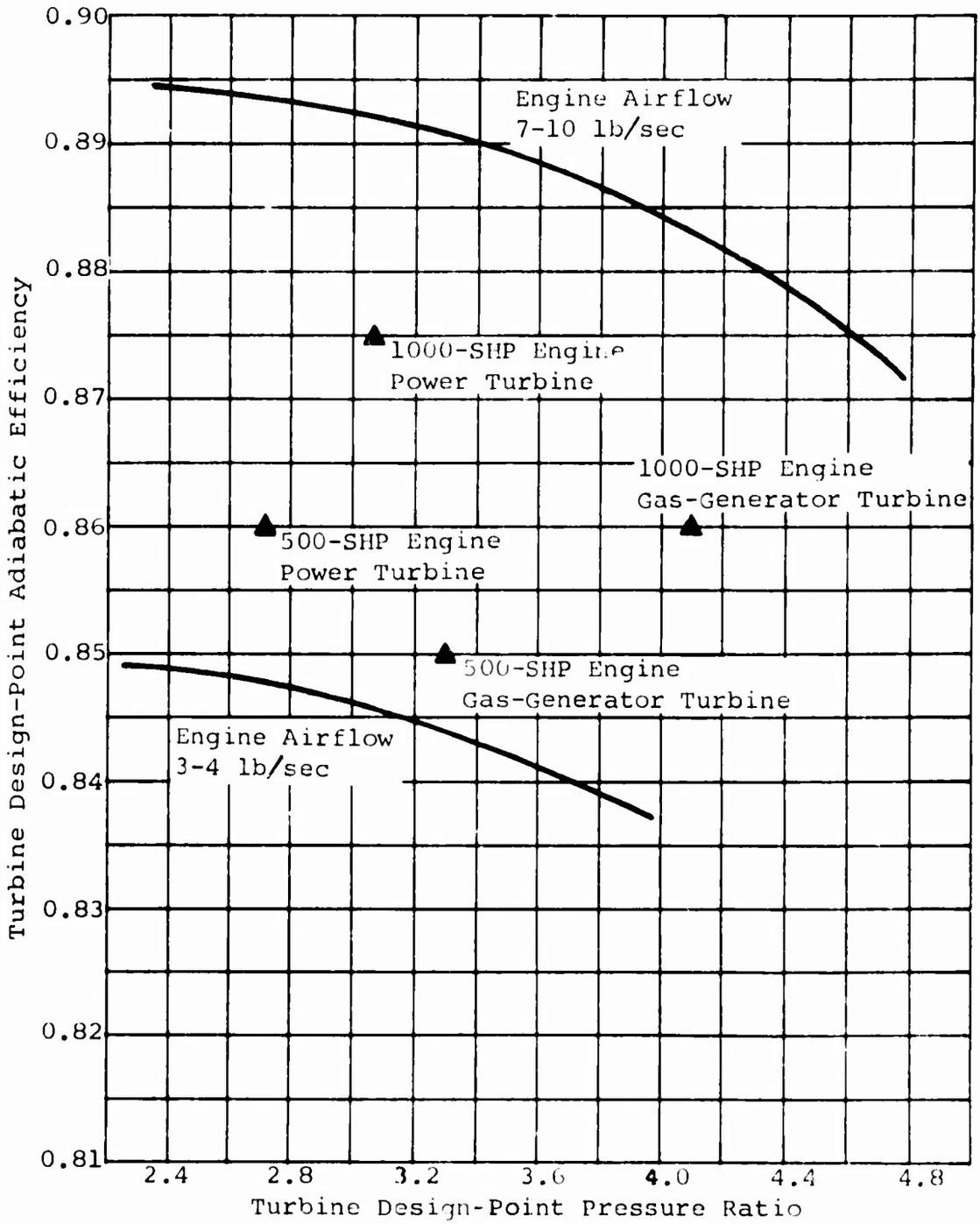


Figure 26. Two-Stage Turbine Design-Point Efficiency Trends for Small Turboshaft Engines Utilizing Available Technologies.

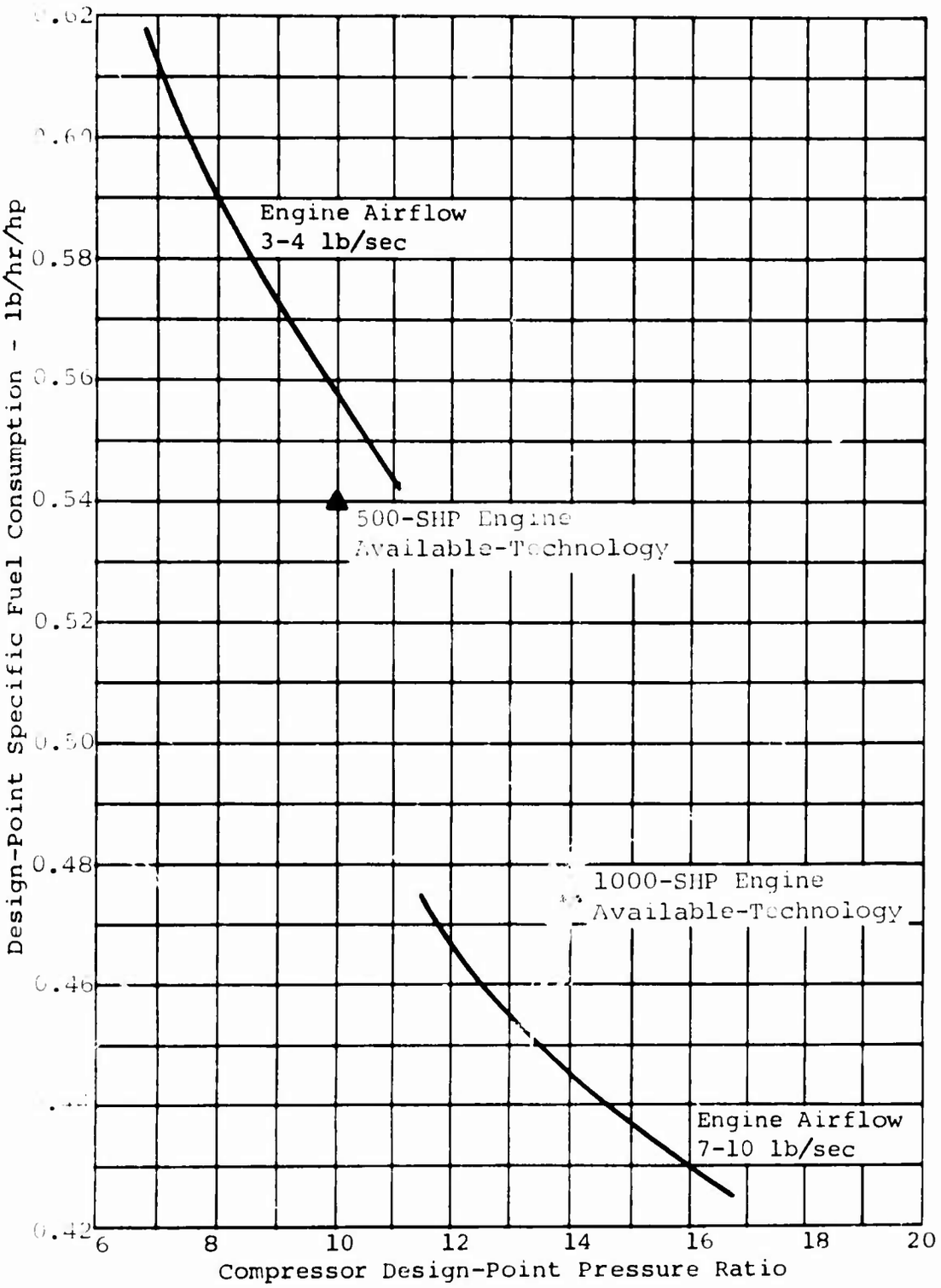


Figure 27. Design-Point Specific Fuel Consumption Trends for Small Turboshaft Engines Utilizing Available Technologies.

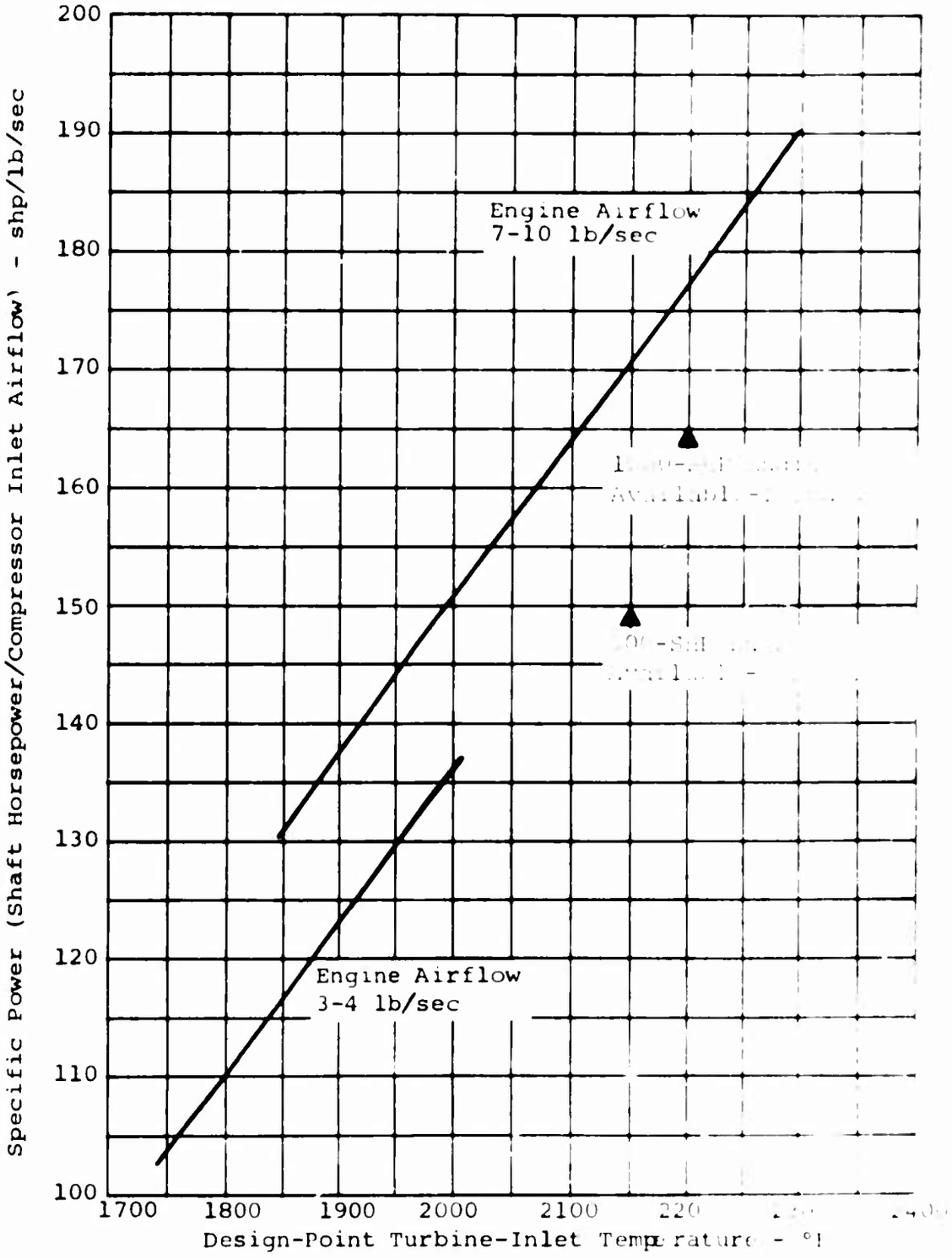


Figure 28. Design-Point Specific Power Trends for Small Turboshaft Engines Utilizing Evolved Technologies.

the design-point efficiency of 0.79, the value in Table III. The power-turbine efficiency trend was similar to the data used previously and was also plotted in Figure 18. The results of the off-design performance calculation are presented in Figure 29, showing referred shaft horsepower and referred fuel flow as functions of referred turbine-inlet temperature for various forward flight speeds. Because these plots are for optimum output shaft speed, the data previously developed in Figures 20 and 21 were used to correct shaft horsepower and SFC for nonoptimum output shaft speed.

As with the earlier-discussed advanced-technology engines, Normal Rated Power was chosen as 85 to 87 percent of the Military Rated Power; but in the case of available-technology engines, the difference in turbine-inlet temperature between the two power settings changed to 125°F. On this basis, turbine-inlet temperature of the available technology turboshaft engine was selected as 2075°F at Normal Rated Power and 2200°F at Military Rated Power.

Engine Configuration

The purpose in defining an engine configuration was to provide dimensional data for installation studies. For the rear-drive arrangement, a very long engine would pose a difficult installation problem. However, the combination of axial and centrifugal compressor stages and the reverse-flow annular burner generally seemed to be favored among the small turboshaft engines under development. Such configurations tend to minimize length of the engine, and the engine outline in Figure 30 was selected as representative for this study.

Engine Weight

Correlations of the dry weights of these small turboshaft engines were developed from basic engine relationships. Basically, engine size and weight are functions of airflow. Specific power (the shaft horsepower developed for each lb/sec of airflow) relates power to airflow, where specific power is primarily a function of turbine-inlet temperature. The relationships of power per pound of airflow and engine weight per pound of airflow combined should permit correlation of the power-to-weight ratio for engines as a function of turbine-inlet temperature (Figure 31). The size of the engines considered, however, has a substantial impact on the correlation. As engines become smaller, limits due to allowable

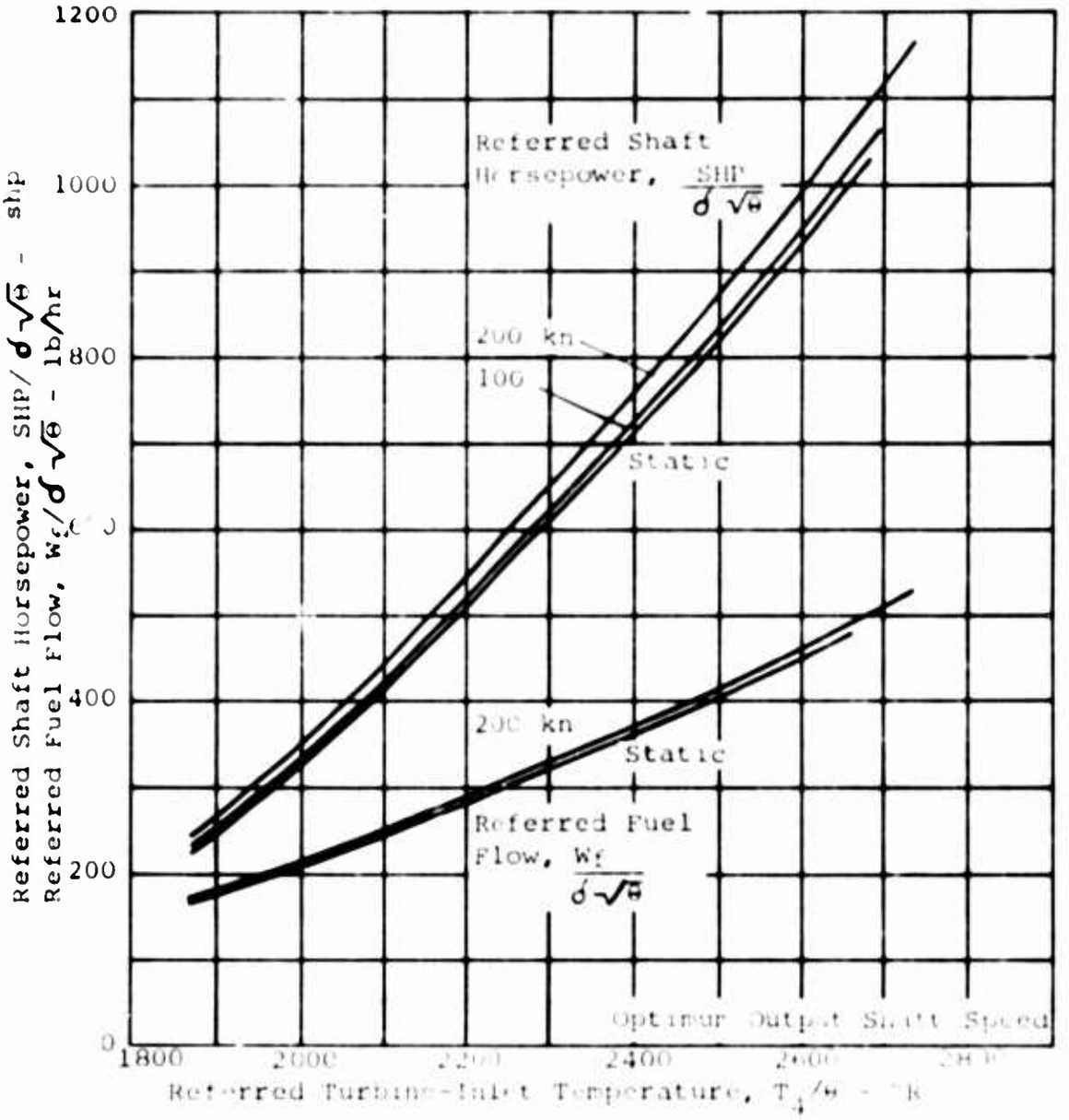


Figure 29. 1000-SHP Available-Technology Simple-Cycle Engine Performance.

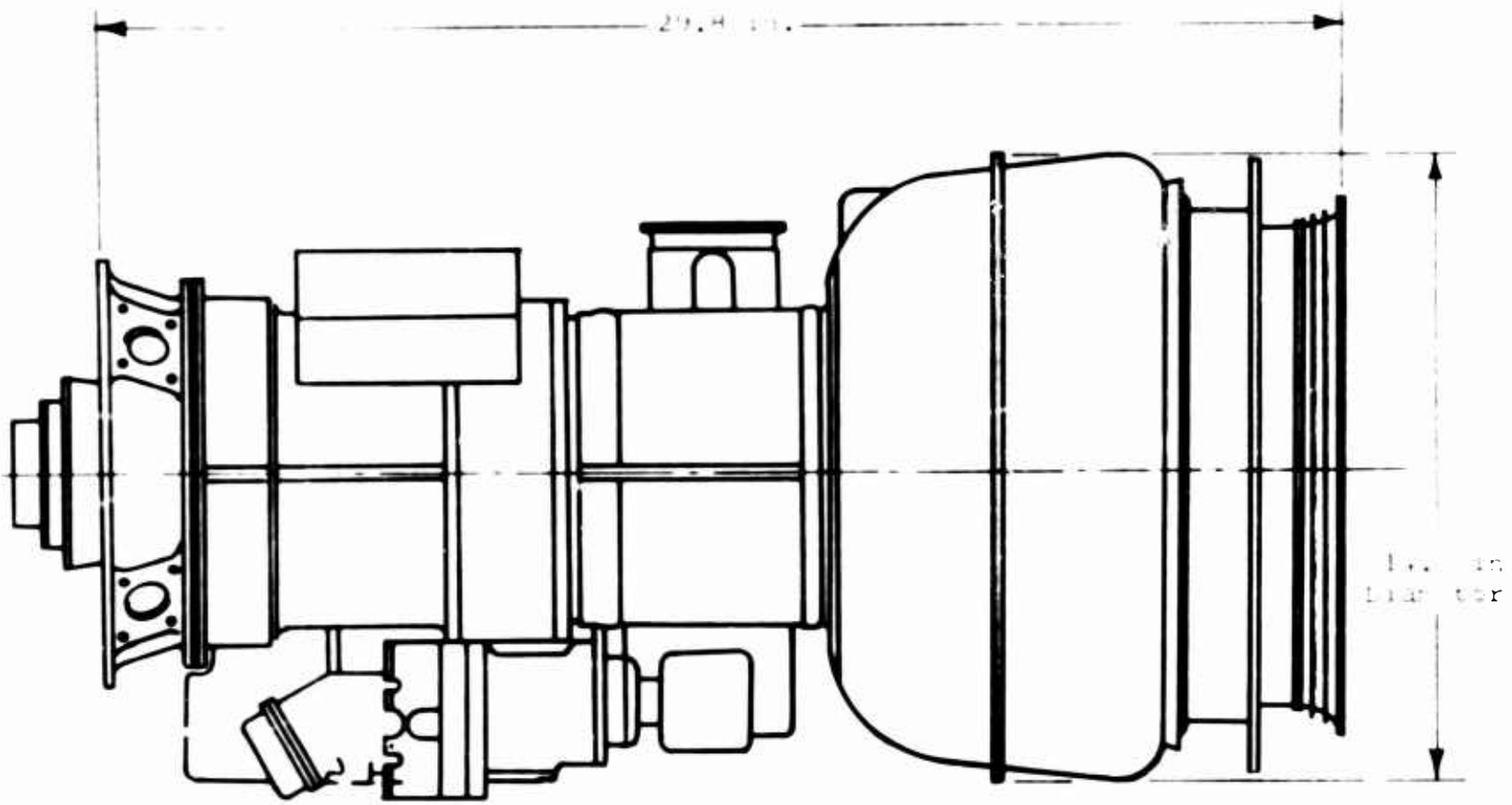


Figure 3. 1000-SHP Available-Technology Simple-Cycle Engine.

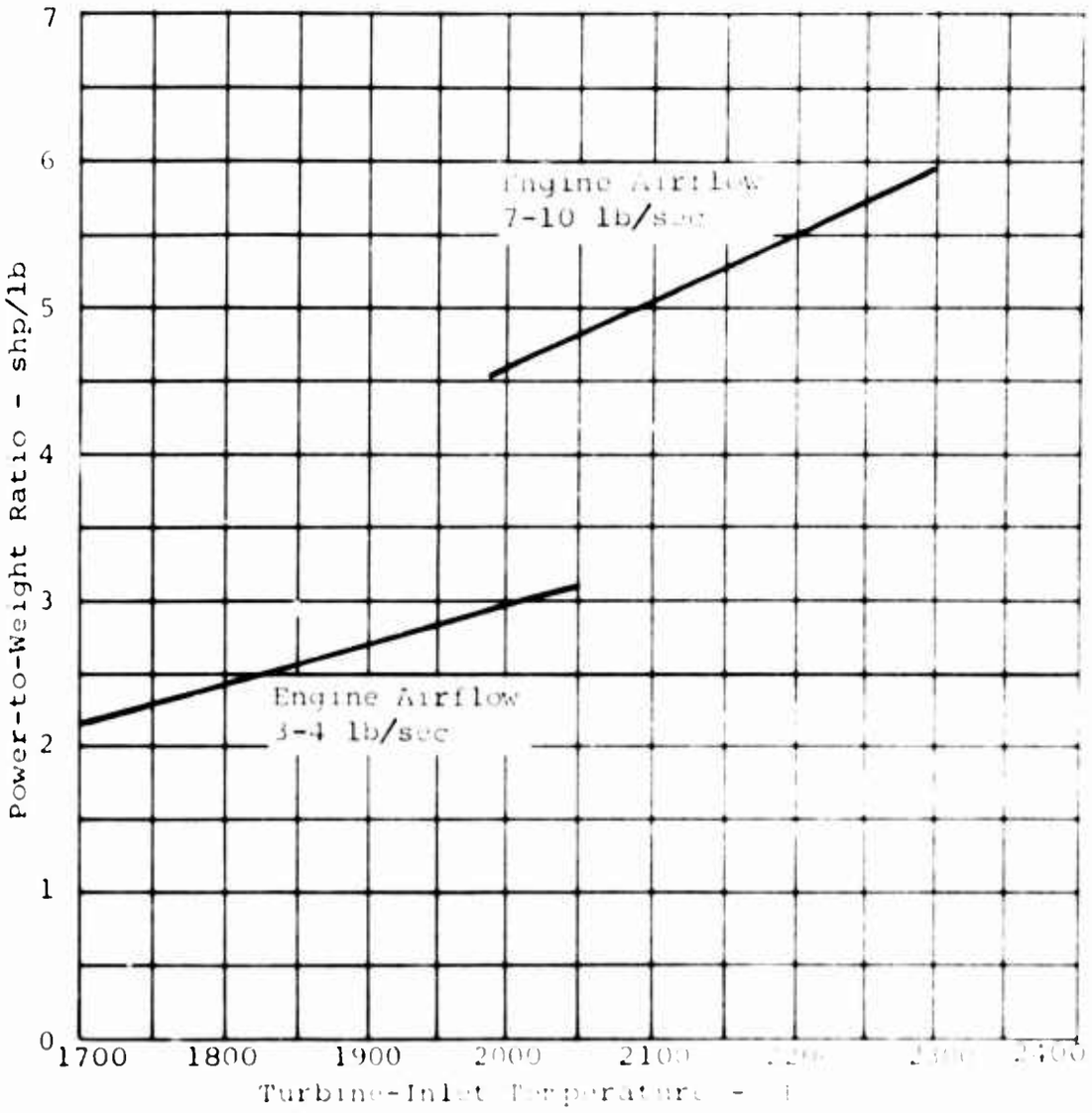


Figure 31. Ratio of Sea Level, [illegible] to Engine Dry Weight [illegible] Engines Utilizing [illegible]

tolerances are encountered for blade chords, material thicknesses, and other dimensions. Limiting bearing dimensions also create a problem in scaling engines to smaller sizes. Tolerances have a greater impact on component performance which can be achieved, and consequently the specific power decreases. As a result, the power-to-weight ratios for engines in the 3- to 4-lb/sec range can be expected to be lower than those for engines in the 7- to 10-lb/sec range.

Of the small turboshaft engines considered, those in the 3- to 4-lb/sec range had approximately 1900°F turbine-inlet temperatures, while those in the 7- to 10-lb/sec range had turbine-inlet temperatures from 2200°F to 2300°F. This interrelationship between size class and turbine temperature, in conjunction with the specific power relationship, suggested a direct correlation of power-to-weight ratio with engine power. In Figure 32, the power-to-weight parameter was correlated solely as a function of the engine rated power. Based on these considerations, 4.15 was chosen as a reasonable value for power-to-weight ratio for a 1000-shp available-technology engine, and was used in subsequent study tasks.

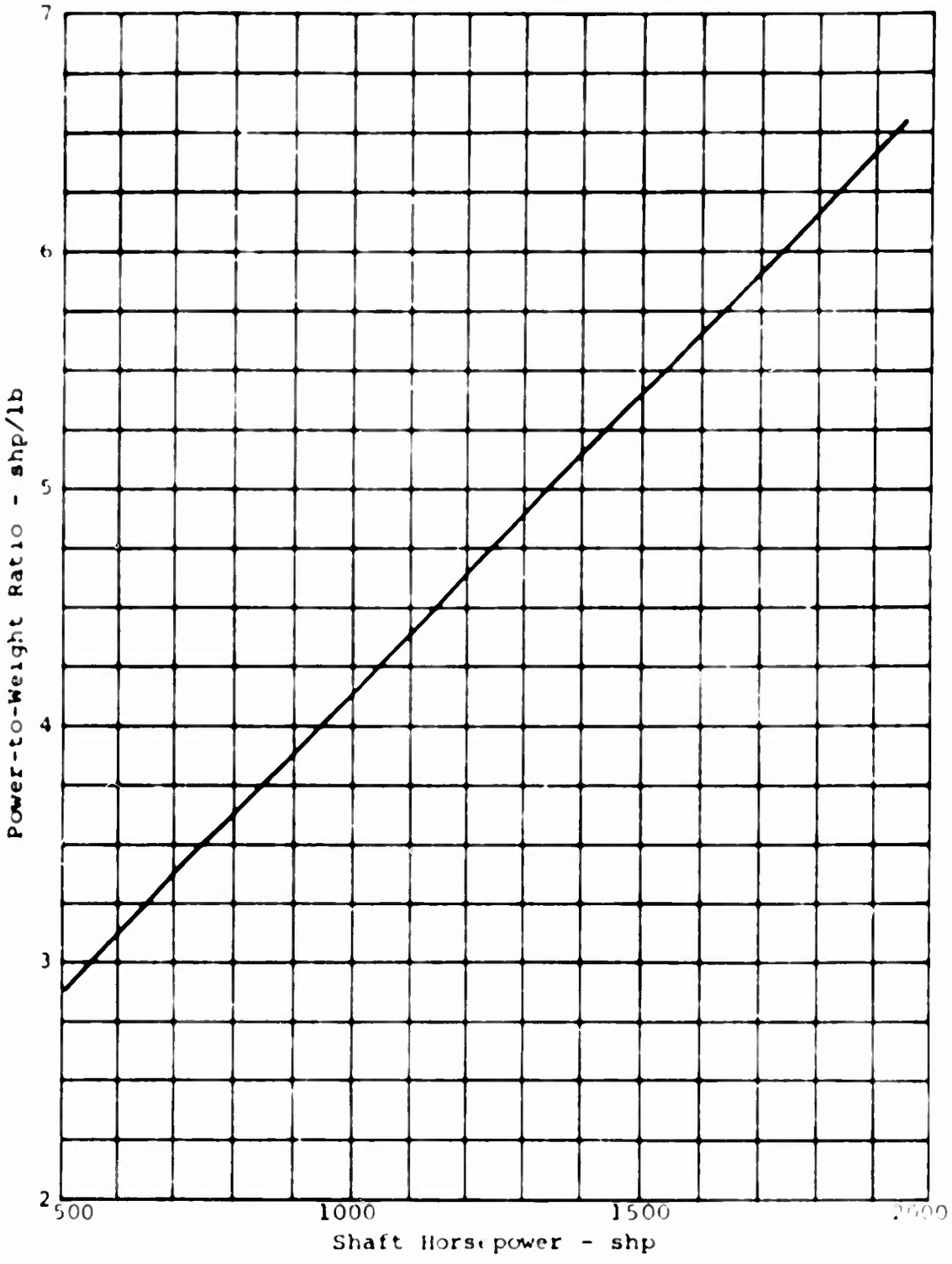


Figure 32. Ratio of Sea Level, 59°F, Shaft Horsepower to Engine Dry Weight for Small Turboshaft Engines Utilizing Available Technologies.

REGENERATIVE ENGINE DESIGN PARAMETERS

The engines in Reference 1 were designed with a 9:1 compressor pressure ratio, very nearly optimum for the regenerative engines. Compared with the simple-cycle engine, the lower pressure ratio for the regenerative engine results in lower compressor-exit temperature, higher turbine-exit temperature, and greater benefit from the recuperator, leading to the desired effect on engine SFC.

For the particular time frame, or state of the art, of this engine (1975 in this instance), the compressor pressure ratio and turbine-inlet temperature are also interrelated with both compressor and turbine configurations. The compressor for the regenerative engines in Reference 1 required only one axial and one radial stage - one axial stage less than the simple-cycle engine. Correspondingly, the gas-generator turbine had only one stage - one less than its counterpart in the non-recuperative engine. These changes would have a favorable impact on engine weight. An additional benefit of the lower pressure ratio was the reduction in cooling required, as illustrated in Figure 1, with the attendant benefits to the cycle performance.

The regenerative engine concepts utilized an annular recuperator of tubular construction wrapped around the turbomachinery, producing a well-integrated engine design which was compact and lightweight. Design-point engines were defined for values of recuperator effectiveness from 0.40 to 0.80, and a range of pressure loss, with configurations for internal and external installation on the aircraft. The output shaft was located at the rear of the engine configurations, consistent with the simple-cycle arrangement discussed in the previous section of this report.

[illegible] maximum benefit in SFC which can be realized with [illegible] recuperator corresponds to the turbine-inlet temperature, it would be advantageous to maintain high temperatures [illegible] a large part of the operating range of the engine. For a selected part-power condition, then, airflow would be less than that of a standard regenerative engine, which would contribute to increased recuperator effectiveness and reduced core pressure loss. For these reasons, variable turbine nozzle vanes in both the gas-generator turbine and power turbine permit regenerative engine operation at high turbine-inlet temperatures over a wide range of part-power conditions,

resulting in improved part-power performance compared with the fixed-geometry regenerative engine. All the data for regenerative engines available in Reference 1, however, was for fixed-geometry turbines. Accordingly, the principal comparisons and assessments reported in this document considered only the fixed-geometry regenerative engines. Studies and assessments of variable-geometry engines are given in Appendix II.

PREVIOUS RESEARCH AND DEVELOPMENT

Various regenerator concepts for aircraft gas turbine engines have been designed during the past decade, and development and test programs have demonstrated the performance potential of regenerators and regenerative engines.

In a contractual effort for USAAMRDL, AVCO Lycoming Division designed a recuperator for the T53 turboshaft engine (Reference 6). In the first phase of this program a multi-wave plate core construction was selected to achieve a compact, lightweight, efficient regenerator. Test cores were fabricated by stamping individual plates and brazing stacked plates in assembly. Effectiveness and pressure loss of the test cores were relatively close to design targets, but leakage problems persisted in the brazed assemblies.

Because of the core leakage problem with this construction, a conventional tube-type recuperator was designed and fabricated in another phase of the program. This concept was a two-pass, cross-counterflow core geometry with compressor air making two passes through the tubes and exhaust gas making one pass outside of and perpendicular to these tubes. The recuperator design was equivalent to a 0.[illegible] effectiveness at the T53-L-9 design point of 1100 shp and 10.7 lb/sec airflow. The recuperator weight was 22[illegible], including [illegible] hardware - the modification kit that was [illegible] on the engine. [illegible] 20.9 lb/hp/sec [illegible].

The Boeing Company [illegible] contract to [illegible] achieving a heat-exchanger [illegible] thermodynamic performance, low pressure [illegible], low weight and volume, and be practical [illegible]. The design concept consisted of rectangular [illegible] small-diameter, thin-walled tubes. [illegible] 0.68 effectiveness at [illegible]

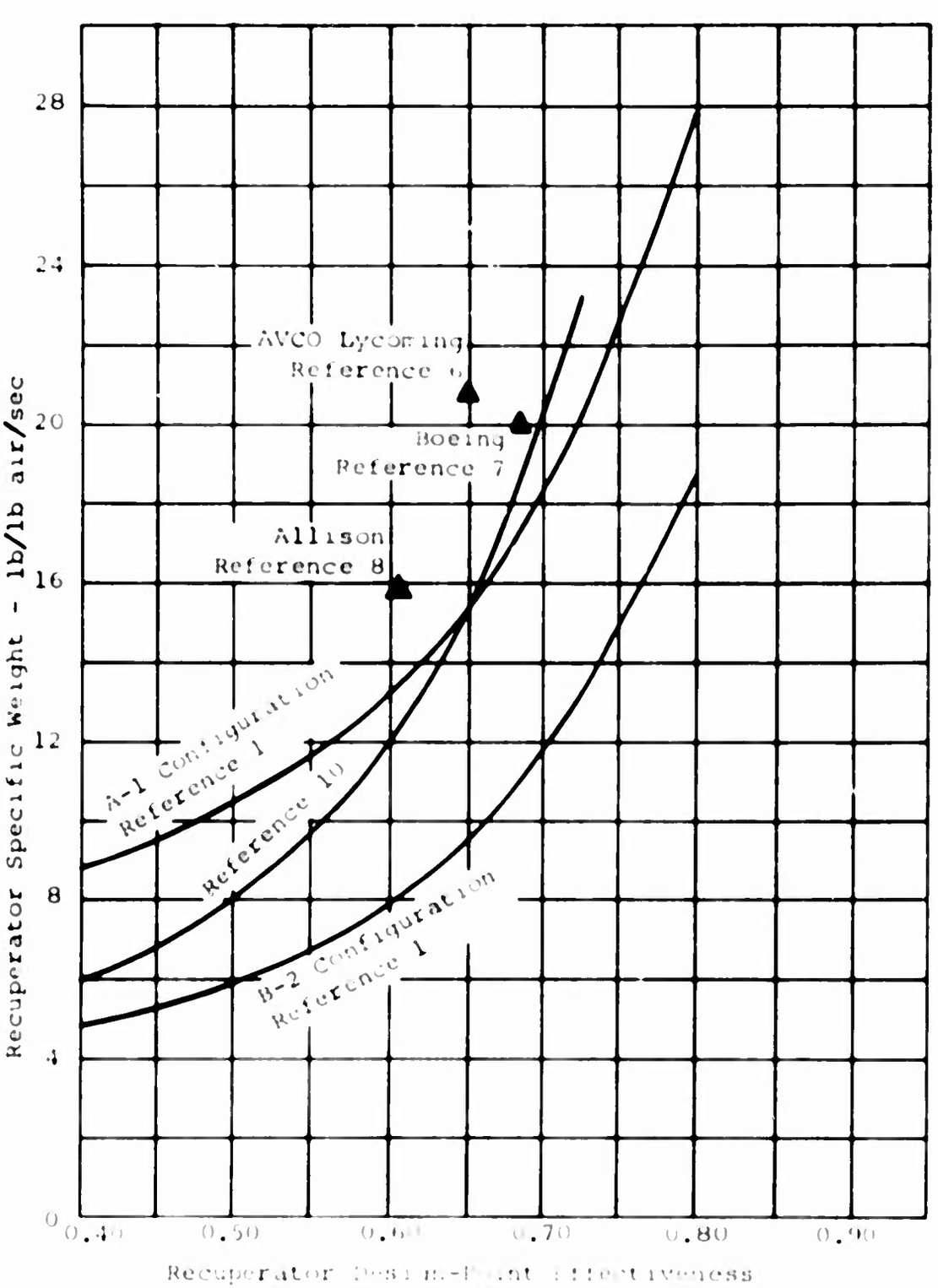


Figure 31. Trends of Recuperator Weight Per Pound of Airflow.

target value for core weight was 10 lb/lb air/sec which was equivalent to a specific weight of 20 lb/lb air/sec for the complete recuperator. This point has been added to the weight correlation in Figure 33.

The program was discontinued at the request of The Boeing Company because of fundamental problems associated with high production costs for the modules of small-diameter tubes, compounded by intergranular corrosion and oxidation in the thin-walled stainless steel material. The program identified the need for further basic research in tube size, candidate materials, and manufacturing techniques. Since that time, encouraging results from follow-on programs indicate that solutions to these problems have been found.

Allison Division of General Motors Corporation performed a flight test program with a regenerative T63 engine in a YOH-6A helicopter (Reference 8). The core geometry of this recuperator was a tubular concept, two-pass cross-counterflow design. The effectiveness achieved on test was 0.607 with an 8.12 percent pressure loss. Recuperator weight was 50.2 pounds, resulting in a specific weight of 15.9 lb/lb/sec, which has been plotted in Figure 33.

For purposes of comparison, the recuperator weights for two of the parametric engine configurations in Reference 1 have been included in Figure 33 - the A-1 configuration designed for external engine installation (in pods, for example) on the aircraft and the B-2 configuration for internal installation. These parametric point designs were configured for 6.0 percent total pressure loss. Weights for the A-1 parametric recuperators should have been and were reasonably consistent with previous trend data, since the designs included external shells, headers, and hardware usually associated with the heat exchanger. The B-2 configurations were significantly lighter, because their unique design eliminated much of the usual recuperator hardware.

AiResearch Manufacturing Division of The Garrett Corporation participated in a 2-year research program for USAAMRDL to determine the hot corrosion resistance and rupture strengths of thin-wall tube materials from which economic lightweight recuperators could be constructed, and to investigate hot corrosion mechanisms at temperatures below 1500°F. Combinations of tube materials and braze materials were tested for periods up to 1000 hours at temperatures between 1100°F and

1500°F in a cyclic hot corrosion test rig. A recuperator fabricated from two such combinations of tube and braze materials was tested in combustion chamber exhaust contaminated with sea-salt, and the hot corrosion mechanisms were studied.

That study program was completed satisfactorily and was used as the basis for design projections given in Reference 1.

ADVANCED-TECHNOLOGY REGENERATIVE ENGINES

AiResearch designed two regenerative engine concepts utilizing an annular recuperator of tubular construction wrapped around the turbomachinery to give compact, lightweight engine packages. The recuperator was considered to be prime structure and acted as the structural backbone of the engine assembly. The engines were sized for approximately 1000 shp - actually the design airflow of each engine was 5.0 lb/sec. The engines could be scaled over a narrow range, however, without change in performance. Two engine concepts were selected: one for external installation on the aircraft (labelled Configuration A-1) and one for internal installation (labelled Configuration B-2).

A matrix of engine design configurations was established covering a range of values of recuperator effectiveness from 0.40 to 0.80 and a range of values of pressure loss from 4 percent to 10 percent. From this matrix of engine configurations, three regenerative engines have been selected for the present study, with recuperator effectiveness of 0.40, 0.65, and 0.80, and pressure loss of 10.0, 6.0, and 4.0 percent, respectively. For the purpose of achieving the lightest possible design, only the B-2 configurations from Reference 1 were used in the present study.

Design-Point Performance

Design-point component data and engine performance parameters were presented for the above three regenerative engines (Table IV). Turbine efficiencies which differed slightly from those quoted in Reference 1 were used to reconcile the engine design-point performance. Primarily, the differences were in turbine efficiency probably because of somewhat different assumptions employed in the thermodynamic calculations. These differences have no appreciable impact on the eventual solution however, since the only purpose in formulating the data was to establish the design-point baseline for the subsequent

TABLE IV. REGENERATIVE ENGINE DESIGN-POINT PARAMETERS (REFERENCE 1)

	Recuperator Effectiveness* .40	.65	.80
Compressor			
Inlet Airflow, lb/sec	5.0	5.0	5.0
Pressure Ratio	9.0	9.0	9.0
Adiabatic Efficiency*	.82	.82	.82
Exit Temperature, °F	601.	601.	601.
Cooling-Air Bleed/Inlet Airflow*	.035	.035	.035
Leakage/Inlet Airflow	.015	.015	.015
Recuperator - Air-Side			
Inlet Flow, lb/sec	4.75	4.75	4.75
Pressure Loss*	.040	.024	.016
Exit Temperature, °F	906.	1081.	1181.
Combustor			
Efficiency*	.99	.99	.99
Fuel/Compressor Inlet Airflow	.0221	.0195	.0179
Pressure Loss*	.03	.03	.03
Gas Generator Turbine			
Inlet Temperature, °F	2300.	2300.	2300.
Inlet Flow, lb/sec	4.861	4.848	4.840
Mechanical Efficiency*	.975	.975	.975
Exit Temperature, °F	1829.	1828.	1827.
Adiabatic Efficiency*	.875	.875	.875
Pressure Ratio	2.55	2.56	2.56
Interstage Turbine Diffuser (3 Percent Cooling-Air Mixed)			
Pressure Loss*	.03	.03	.03
Power Turbine			
Inlet Temperature, °F	1806.	1805.	1804.
Inlet Flow, lb/sec	5.011	4.998	4.990
Exit Temperature, °F	1345.	1329.	1320.
Adiabatic Efficiency*	.91	.91	.91
Pressure Ratio	2.88	2.99	3.05
Exhaust Diffuser			
Pressure Ratio	1.04	1.04	1.04
Recuperator - Gas-Side			
Inlet Flow, lb/sec	5.011	4.998	4.990
Pressure Loss*	.06	.036	.024
Exit Temperature, °F	1068.	900.	794.
Specific Power, hp/lb/sec	187.3	192.7	195.4
Shaft Power, hp	937.	964.	977.
SFC, lb/hr/hp	.425	.364	.331

*Efficiencies, effectiveness, pressure losses, bleed and leakage flows expressed as decimal fractions.

off-design calculations. The shaft horsepower and SFC for the regenerative engines were in agreement with the data in Reference 1.

Off-Design Performance

Off-design performance was developed for the three regenerative engines to provide data for the spectrum of altitude-ambient temperature conditions, flight speeds, and aircraft power requirements. The assumed compressor performance map for the regenerative engines was plotted in Figure 34, with the efficiency trend along the engine operating line. Gas-generator turbine efficiency was assumed constant along its operating line. The power turbine efficiency trend at optimum output-shaft speed (Figure 35) was calculated from typical two-stage turbine performance characteristics. The decrease in engine airflow at part-power operating conditions resulted in an increase in recuperator effectiveness and a decrease in total pressure drop. These changes in recuperator part-power performance were plotted in Figures 36 and 37.

Results of the off-design performance calculations were plotted in Figures 38, 39, and 40 in the format of referred shaft horsepower and referred fuel flow as a function of referred turbine-inlet temperature for various flight speeds - at optimum output shaft speed.

Corrections for nonoptimum output shaft speed were the same as in the case of the simple-cycle engines, Figures 20 and 21. The turbine-inlet temperatures of the regenerative engines were selected as 2150°F at Normal Rated Power and 2300°F at Military Rated Power, consistent with the advanced-technology simple-cycle configuration.

Engine Configuration

A cross-section drawing of the 0.65 effectiveness regenerative engine is given in Figure 41. The turbomachinery flow path would be identical for all three values of effectiveness, the only difference being the dimensions of the annular recuperator. Referring to Table I, engine length with the 0.40 effectiveness recuperator would be the same as shown in Figure 41 for the 0.65 effectiveness, but the longer tube length for the 0.80 effectiveness recuperator would result in a longer overall engine length.

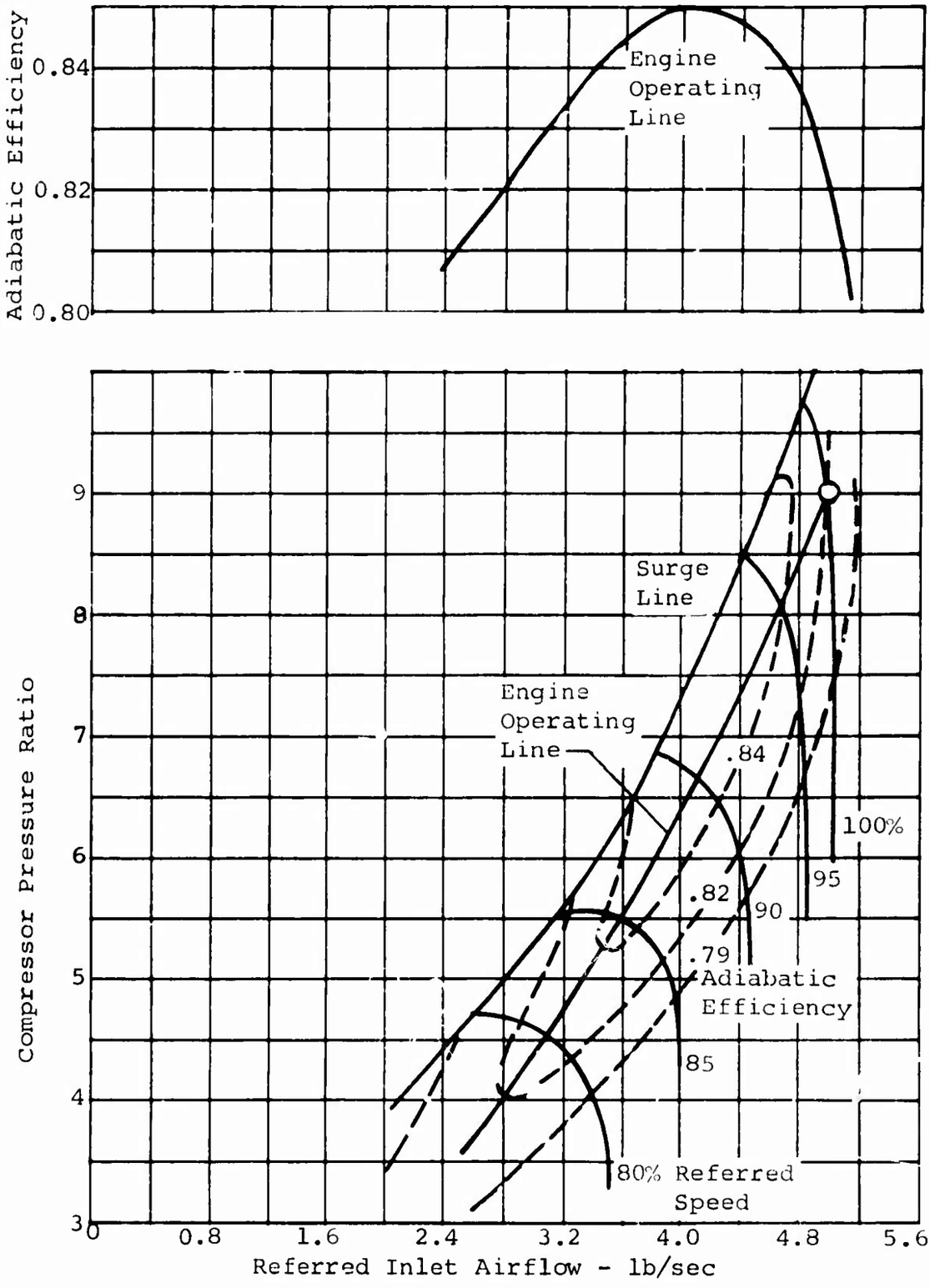


Figure 34. Compressor Performance Map for Advanced-Technology 1000-SHP Regenerative Engines.

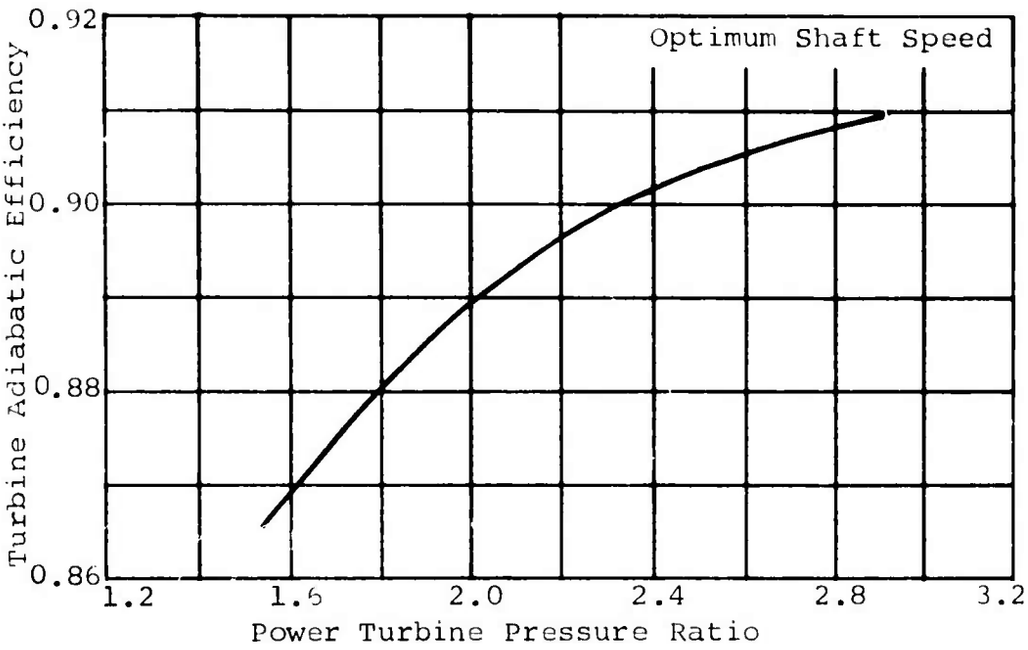


Figure 35. Power Turbine Off-Design Efficiency Trend for 1000-SHP Advanced-Technology Regenerative Engine.

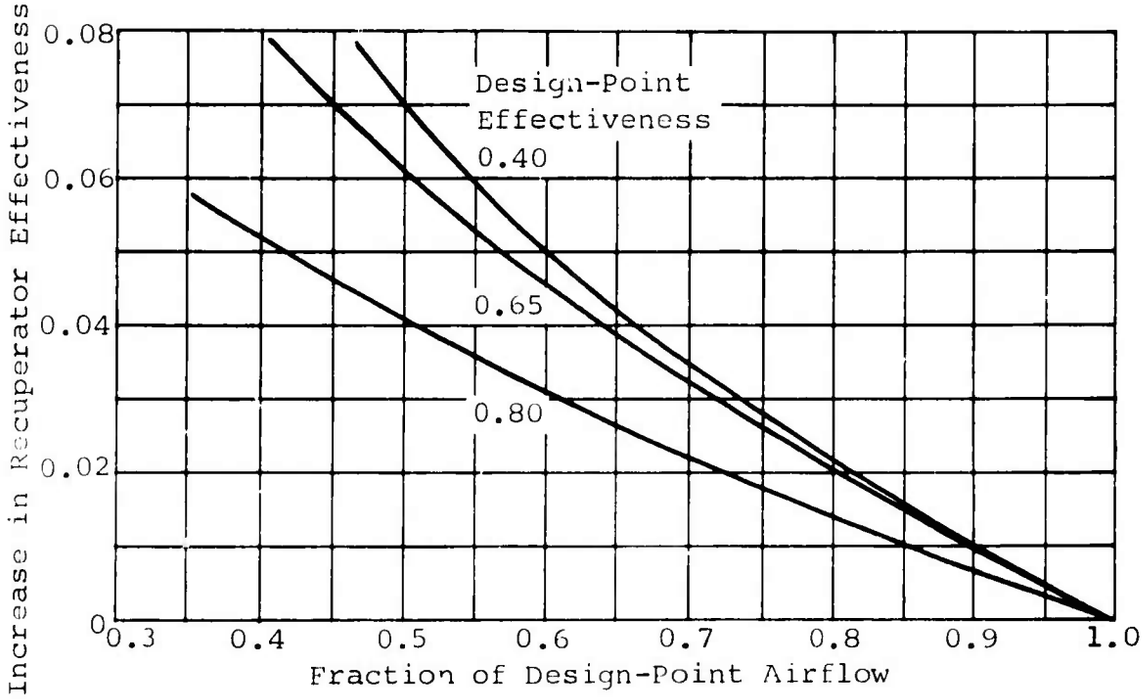


Figure 36. Increase in Recuperator Effectiveness at Part-Power Flows.

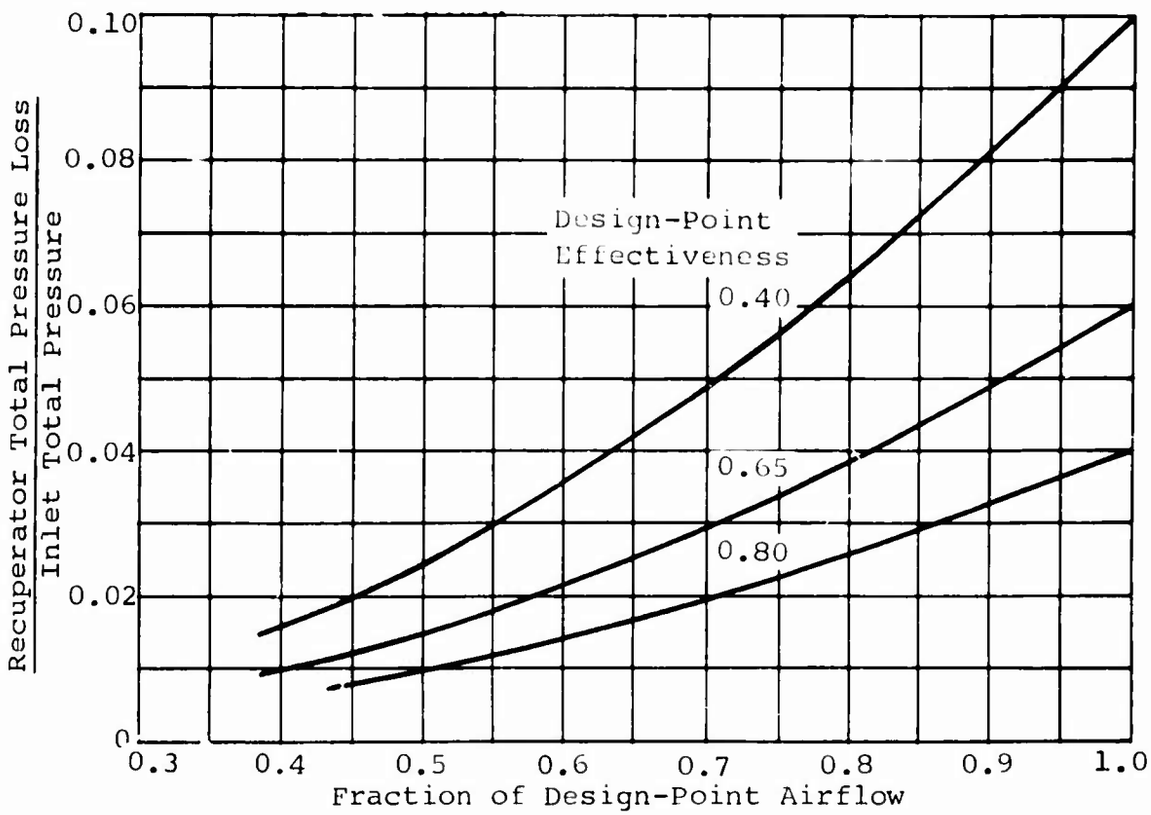


Figure 37. Decrease in Recuperator Pressure Loss at Part-Power Flows.

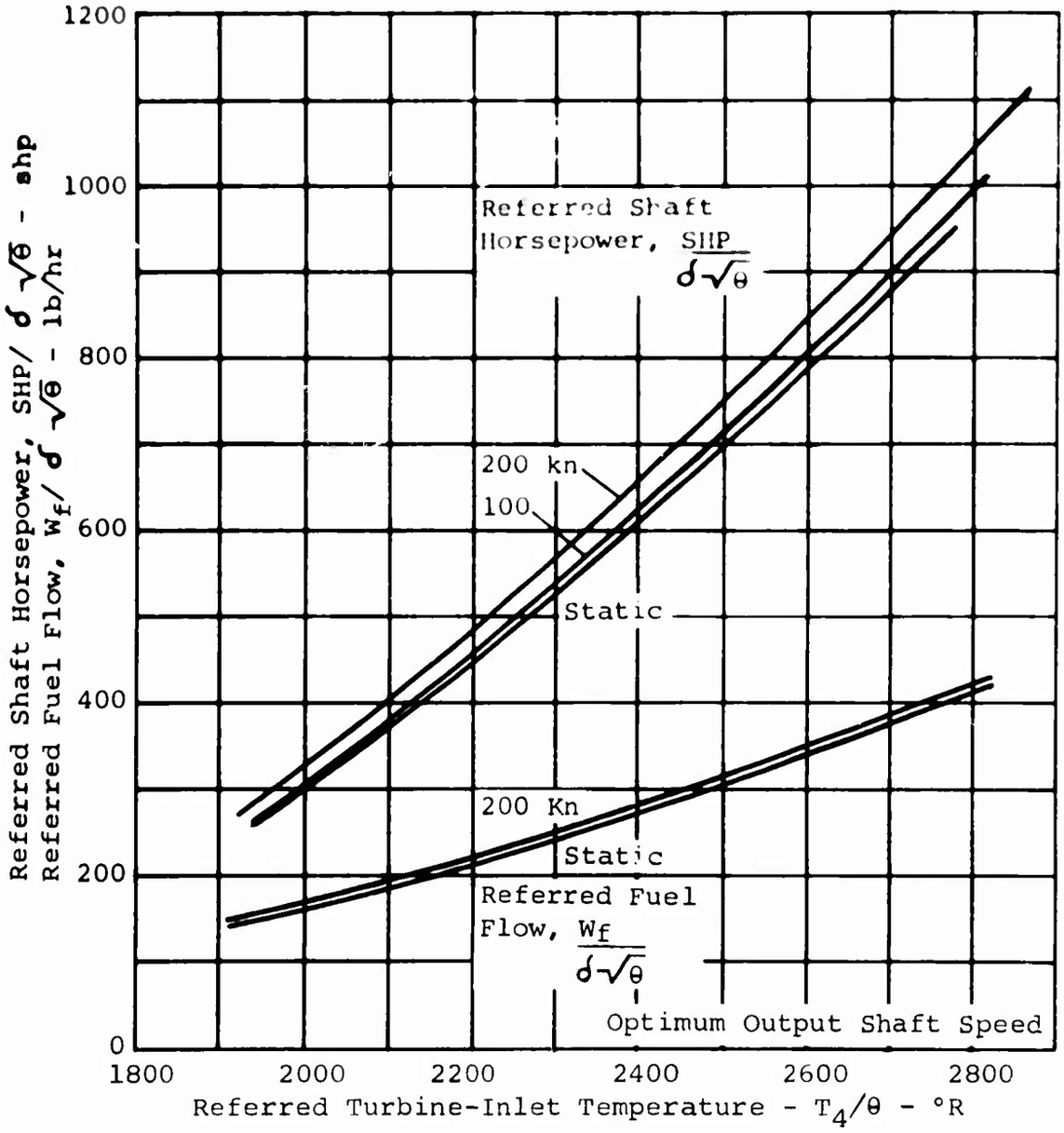


Figure 38. 1000-SHP Advanced-Technology Regenerative Engine Performance (Design-Point Effectiveness = 0.40).

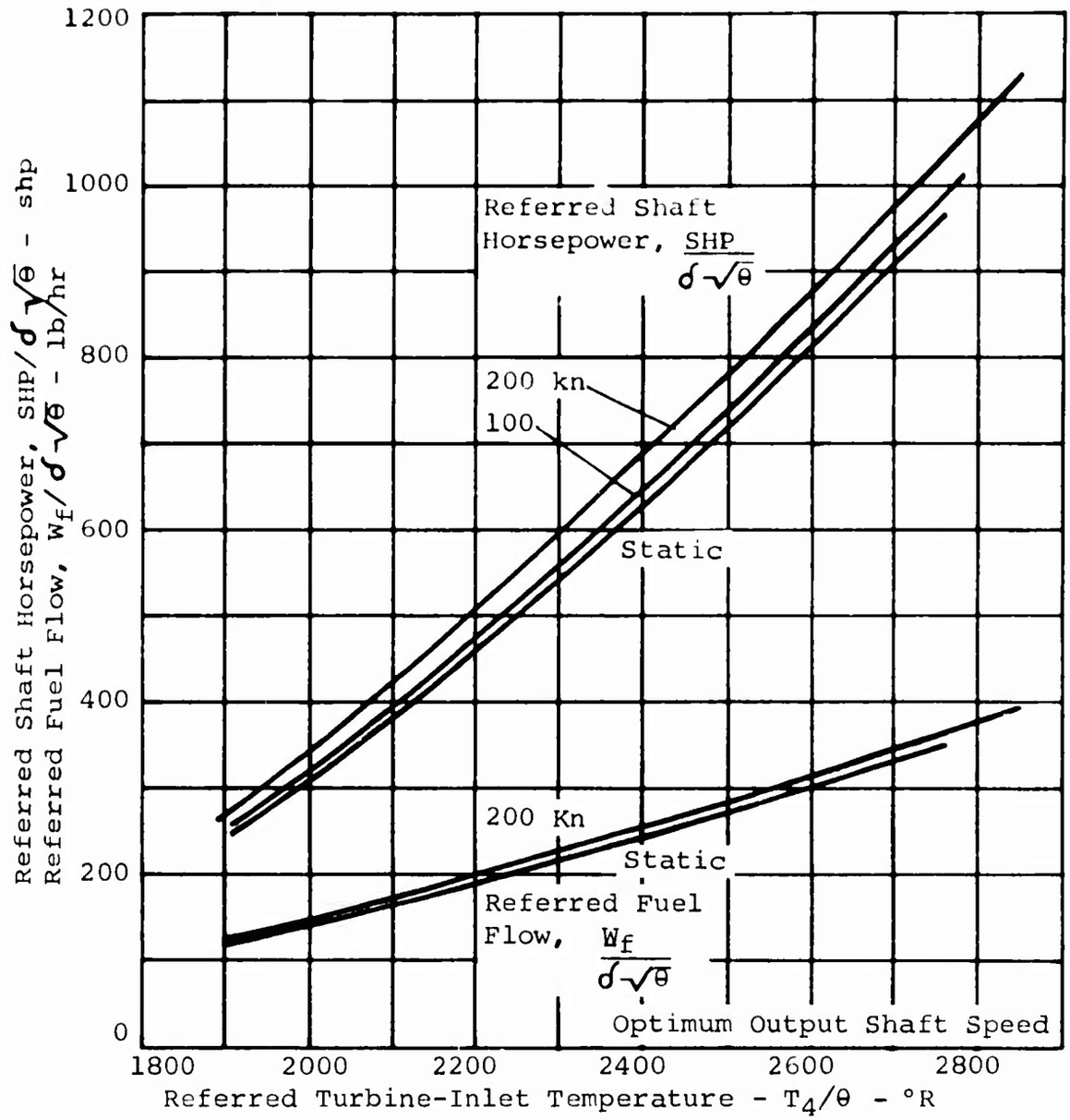


Figure 39. 1000-SHP Advanced-Technology Regenerative Engine Performance (Design-Point Effectiveness = 0.65).

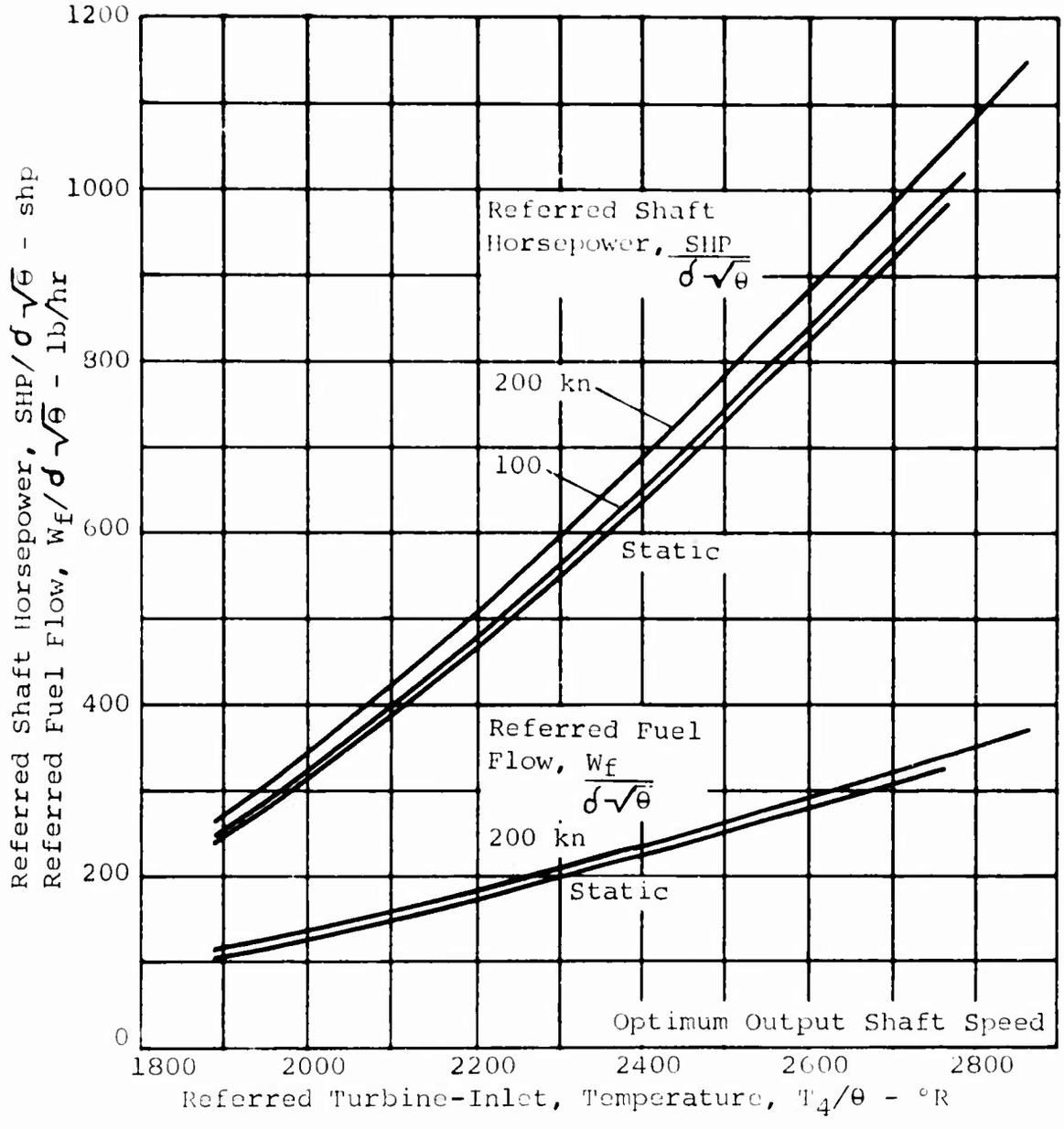


Figure 40. 1000-SHP Advanced-Technology Regenerative Engine Performance (Design-Point Effectiveness = 0.80).

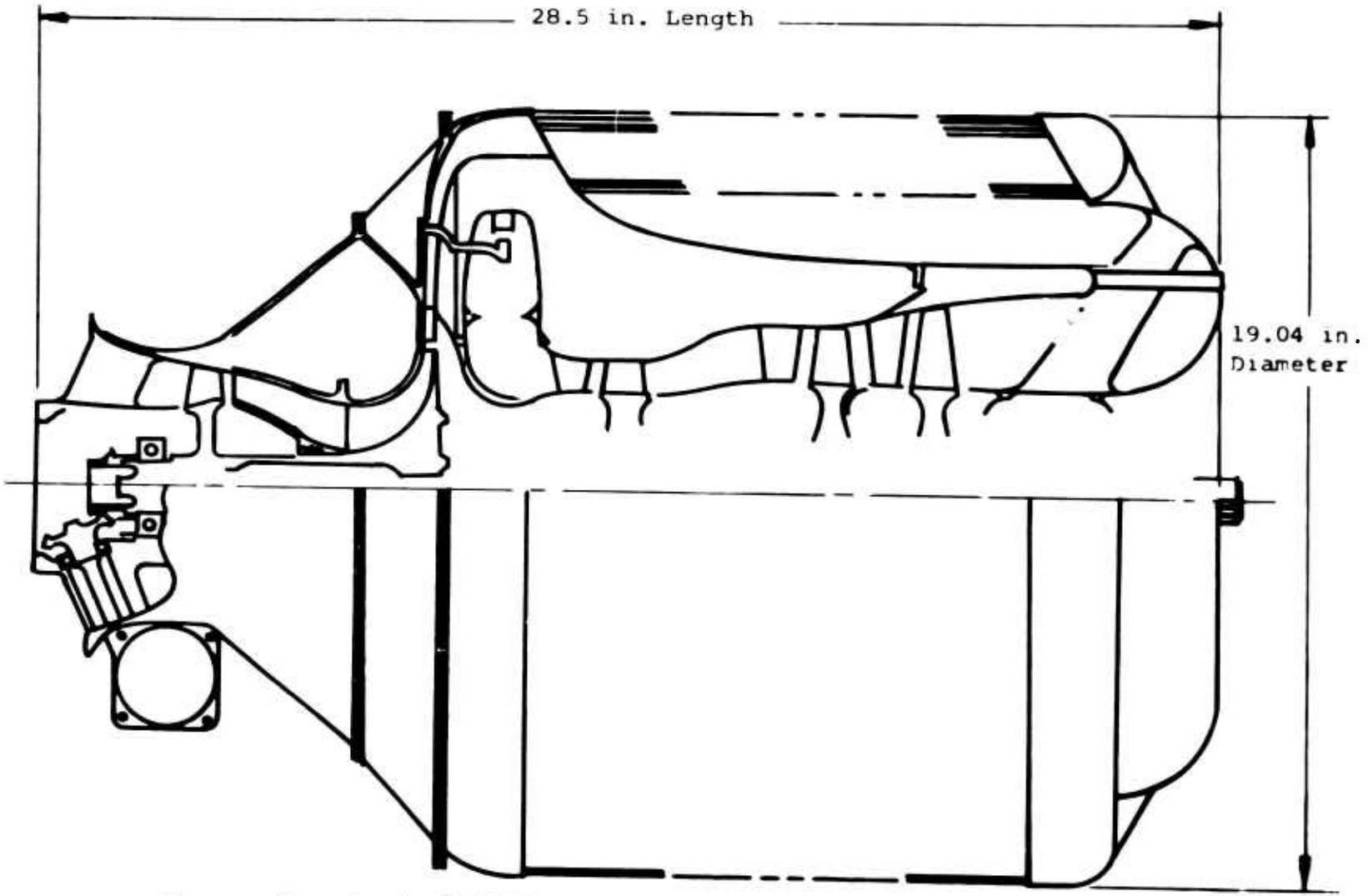


Figure 41. Nominal 1000-SHP Advanced-Technology Regenerative Engine, 0.65 Effectiveness (Reference 1).

Engine Weight

Regenerative engine weights were scaled linearly in proportion to power over the narrow range required to normalize data to the 1000-shp size, using the data in Table I, and dry weights for regenerative and nonregenerative 1000-shp engines were plotted in Figure 42 to be used in the aircraft conceptual designs. In Figure 42 the rapid increase in regenerative engine weight at high values of effectiveness was readily apparent. This engine weight increase more than offset decreased fuel requirements resulting from improvements in SFC as effectiveness approached 0.80. Consequently, the optimum recuperator from an aircraft gross weight standpoint had an effectiveness considerably less than 0.80.

Engine Reliability, Maintainability

Regenerative engine reliability and maintainability data were provided by AiResearch and have been included with data for the other aircraft subsystems in the Aircraft Comparative Analyses section of this report.

Engine Cost

Development costs for simple-cycle engines were correlated in Figure 23. Data provided by AiResearch indicated that development costs for regenerative engines would be 20 percent higher than those for simple-cycle engines. This increase should reflect costs for design, prototype (and production) tooling, material, fabrication, assembly, and component testing applicable only to the recuperator, since such costs for other engine components as well as requirements for engine endurance testing to achieve qualification would be the same for both regenerative and nonregenerative engines.

Procurement costs for turboshaft engines were plotted in Figure 24. The slope of these trend curves should be applicable to both regenerative and nonregenerative engines. Reference 1 provided the cost data for the engine configurations in Table IV, which were plotted as point data in Figure 24 with the trend lines to scale the cost data to larger engine sizes.

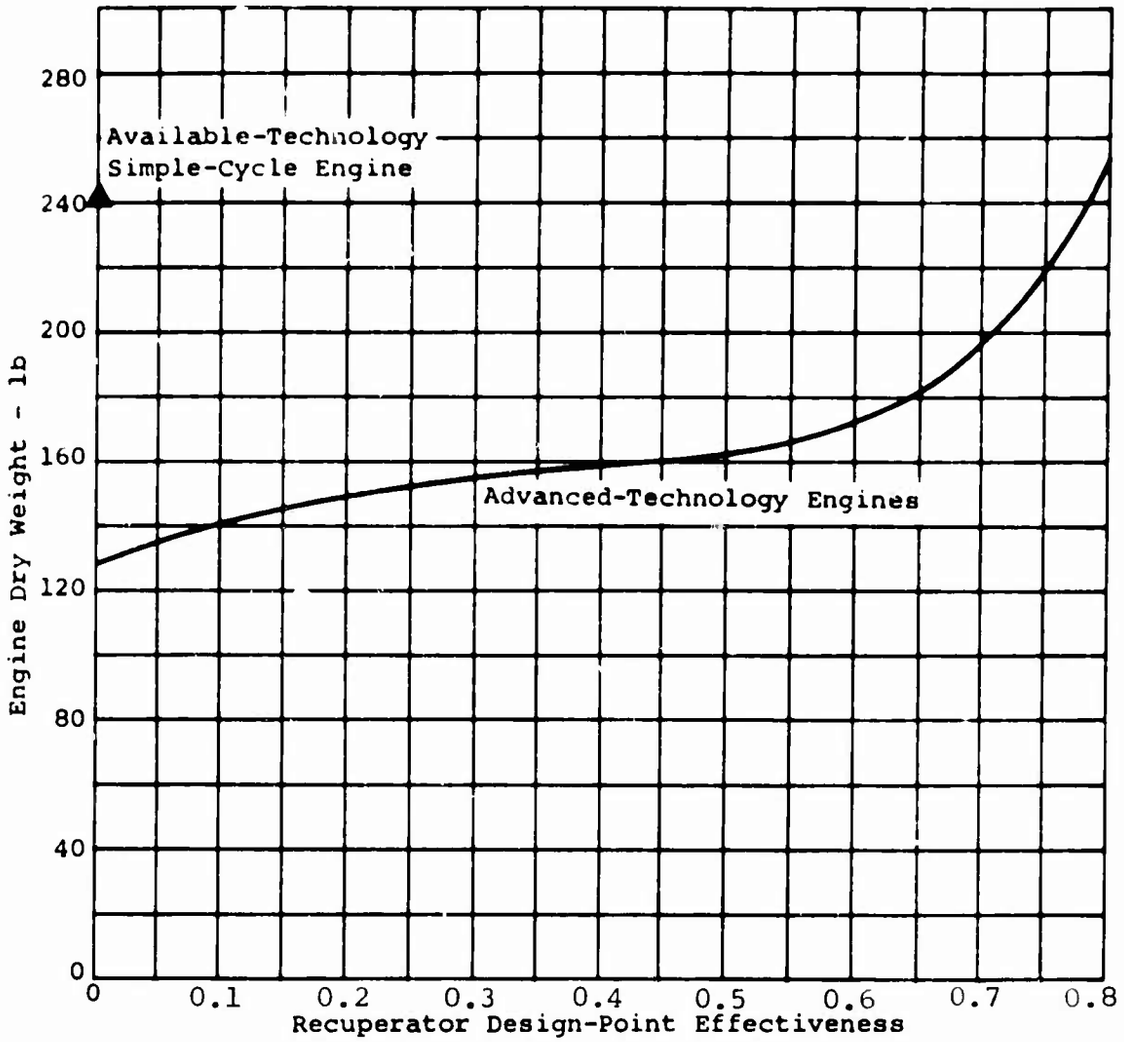


Figure 42. Dry Weights for 1000-SHP Regenerative and Non-regenerative Engines.

AIRCRAFT CONFIGURATION STUDIES

A conceptual design of an aircraft was required in which each propulsion system could be installed and evaluated in comparison with the other candidate systems. Boeing performed sufficient vehicle advanced design to define the characteristics of a utility transport helicopter which could enter production in 1975. Primary and secondary missions defined for this aircraft in the AIRCRAFT MISSION DEFINITION section generally established payload and range requirements, cargo compartment size, and crew accommodations.

BASELINE AIRCRAFT

An aircraft configuration was outlined with installed power in the 1000-shp class, suitable for the utility mission and payload, which was selected as 1200 pounds. It was recognized that to achieve the design payload of 1200 pounds, however, somewhat more than 1000 shp might be required. Engine performance previously developed would be assumed unchanged over a narrow scaling range, but engine weight would be scaled as a function of required shaft horsepower. A baseline helicopter is pictured in Figure 43. A single main rotor plus tail rotor arrangement was selected as the most practical system for a utility transport aircraft of this size. The design had a high tail boom, and sliding cargo compartment doors were incorporated to facilitate the loading of cargo or litter patients. The cargo compartment provided removable passenger accommodations for four to five people, and space was provided for a pilot, copilot, and gunner. The high tail boom, in-line dynamic system components, and flat-deck engine mounting arrangement were well suited to evaluate alternative engine concepts and installations.

A 40-foot-diameter growth version of the Boeing BO-105 four-bladed, rigid-rotor system was selected as the appropriate size for the study aircraft, in addition to representing 1975 helicopter technology. Tail rotor diameter was sized to give ample yaw control in all modes of flight, plus main rotor torque compensation.

Other features of the baseline vehicle design are:

1. Open access between cargo compartment and cockpit

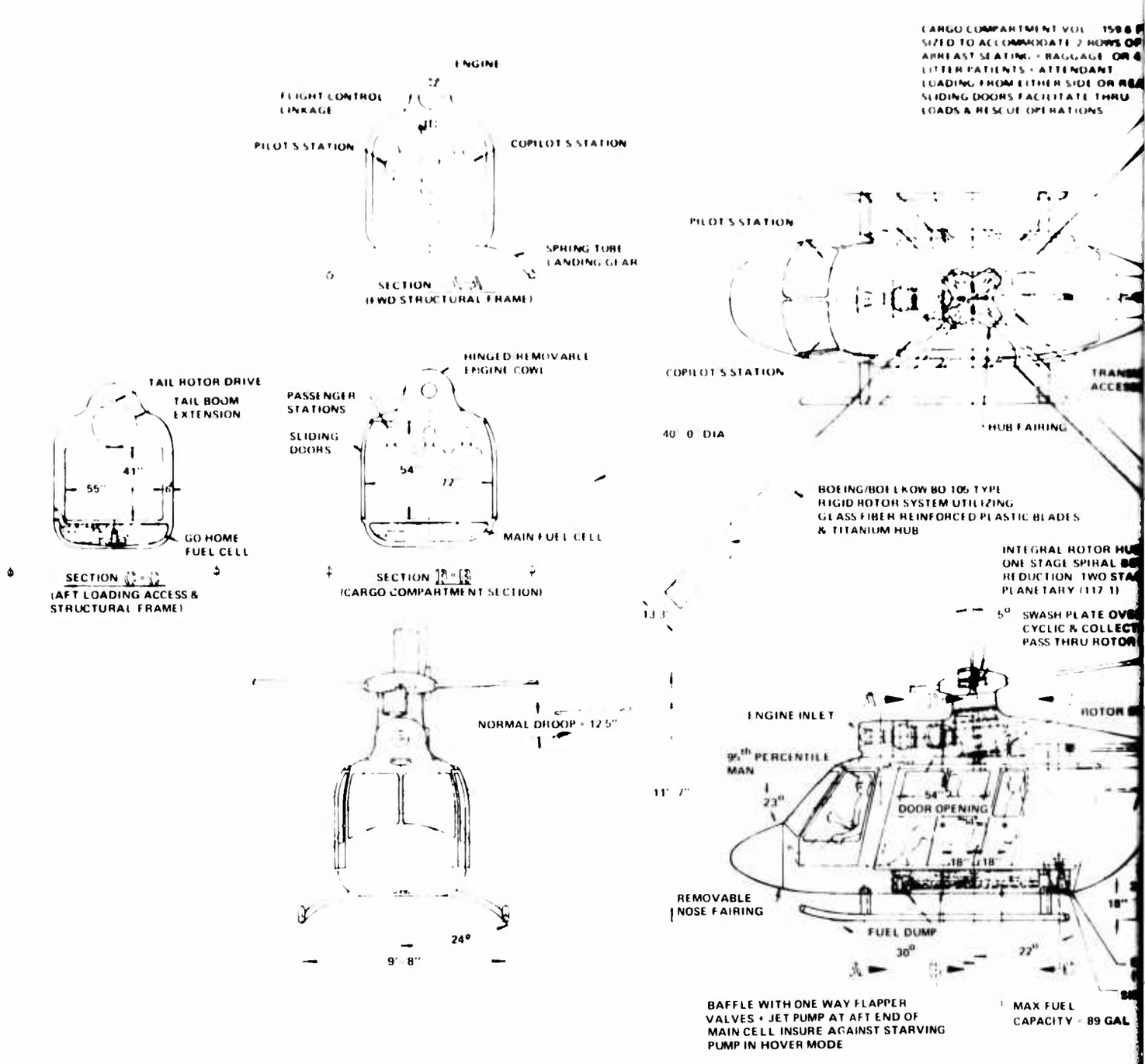
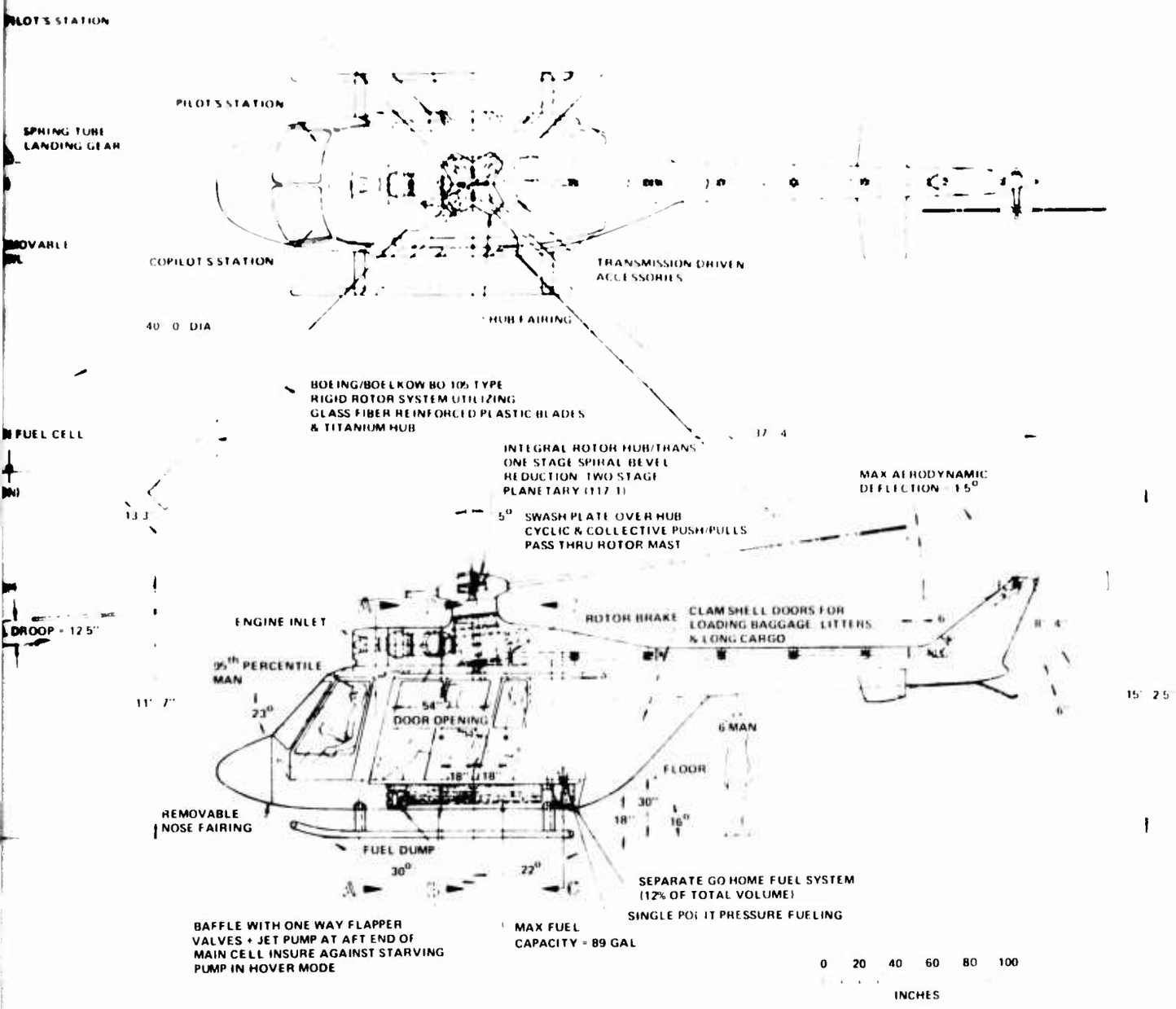


Figure 43. General Arrangement Drawing of Baseline Utility Transport Helicopter.



ing of Baseline
pter.

2. Belly-mounted self-sealing fuel cell, with "go-home" separate fuel system

3. Single-point pressure fueling

4. Low level of cargo deck with loading possible from three sides

5. Dynamic components located above, and protected by, basic structure and main fuel cell

6. Tail rotor above the heads of ground personnel and providing more than adequate landing flare angle

7. Adequate ground-to-belly clearance for rough field operation

8. Integrated main rotor and power transmission system

9. Main rotor brake, foldable blades, and removable conical section tail boom

Other considerations in the basic design aimed at minimizing cost were:

1. External sliding cargo doors

2. Tail rotor drive divided into equal-length, interchangeable, segments

3. Simplified spring-tube landing gear

4. Fuselage mold lines incorporating a minimum of compound curvature

ALTERNATIVE CONCEPTUAL DESIGNS

Internally-mounted single-engine propulsion systems and internally- and externally-mounted twin-engine propulsion systems, with regenerative and nonregenerative engines, were investigated. Because twin-engine aircraft could offer increased mission reliability, these configurations were included among the conceptual designs generated.

The advanced-technology engine designs presented in Reference 1 all had a rear-drive output shaft. Therefore, the engines were located forward of the main transmission in the aircraft with the internally-mounted single engines and twin engines. For the twin-engine aircraft with external mounting, the engines were oriented laterally, driving inboard into the sides of the main transmission. In this concept, the engines were mounted in wing-like structures - the intakes were outboard facing forward and turned the inlet air 90° to the engine inlet flange, while the exhaust turned the hot gas 90° in the aft direction. A preliminary integration of propulsion system arrangements and basic airframe was accomplished, including a cursory investigation of engine air intake and exhaust systems, engine cooling, and accessories, to determine which concepts should be retained for further consideration.

Comparative weight and performance analyses were conducted for each of six conceptual designs with the following propulsion systems:

1. Single-engine, regenerative, internally-mounted
2. Single-engine, nonregenerative, internally-mounted
3. Twin-engine, regenerative, internally-mounted
4. Twin-engine, nonregenerative, internally-mounted
5. Twin-engine, regenerative, externally-mounted
6. Twin-engine, nonregenerative, externally-mounted

These six concepts were reduced to three possible options to assess the impact of the engine installation on the aircraft design. The three options are pictured in Figure 44. A ranking system was devised to compare the three options, and the criteria considered are listed in Table V. An "X" was used to mark the best propulsion system option for each of these selected criteria.

The single-engine designs proved to be the best, or equivalent to the best, for all but a few of the evaluation criteria. The conclusion drawn from this ranking exercise was that the single-engine aircraft was the best configuration for a helicopter performing a utility transport mission. The single-engine aircraft were lighter, had a higher payload-to-empty

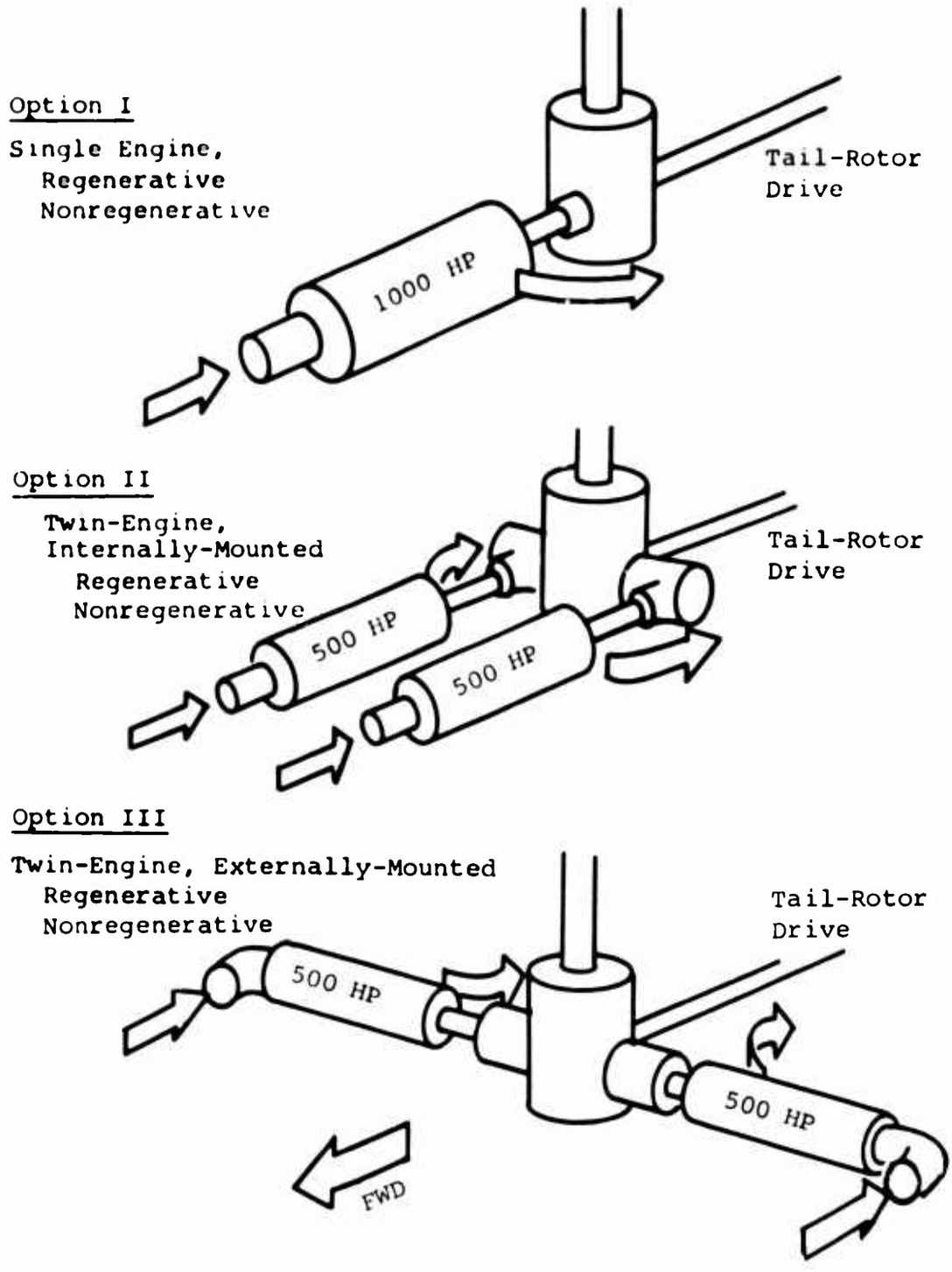


Figure 44. Alternative Engine Installations for Single-Engine and Twin-Engine Utility Helicopters.

TABLE V. RANKING CHART FOR ASSESSMENT OF SINGLE-ENGINE VERSUS TWIN-ENGINE UTILITY HELICOPTERS

Evaluation Criteria	Option I	Option II	Option III
Primary Importance			
Aircraft Configuration			
Gross Weight	x		
Payload/Empty Weight	x		
Aircraft System			
Emergency (O.E.I.) Capability		x	
Reliability-Availability	x		
Cost/Producibility	x		
Secondary Importance			
Powerplant			
Engine Performance	x	x	
Power Management	x		
Maintainability	x		
Drive System			
No. of Gearboxes	x	x	x
No. of Gears, Drive Efficiency	x		
Drive System Weight	x		
Aircraft Configuration			
Design Complexity (Subsystems)	x		
Drag, Forward Flight	x	x	
Download	x	x	
Aircraft System			
Vulnerability	x	x	
Flight Safety/Survivability			x
Minor Importance			
Powerplant			
Separator Integration		x	
Potential FOD Problem	x		
IR Suppressor Integration			x
Engine Interchangeability	x		
Growth Potential	x		
Aircraft Configuration			
Fuselage Structural Interface	x		
Firewall Requirements			x
Powerplant Cooling Problems	x		
Aircraft Center of Gravity	x	x	x
Stability Effects	x	x	x
Aircraft System			
Ground Safety(Inlet/Exhaust)	x	x	x
Transportability	x		

weight ratio, were more reliable and available, were less expensive and easier to maintain, and showed less mechanical complexity. The relative importance of engine-inoperative capability would be the only factor leading toward twin-engine helicopters. Therefore, single-engine aircraft were retained for subsequent study tasks, with propulsion systems including:

1. A simple-cycle advanced-technology engine

2. A simple-cycle available-technology engine

3. Regenerative advanced-technology engines with recuperators having three different values of effectiveness

Data pertaining to the twin-engine designs are given in Appendix I. The subsequent tasks involved comparisons of regenerative and nonregenerative engines in the aircraft installation - comparisons which would be equally valid for either single-engine or twin-engine helicopters.

PARAMETRIC PERFORMANCE, WEIGHT STUDIES

Parametric performance and weight data were developed for each of the five aircraft design concepts just enumerated with five different single-engine propulsion systems. Three different parametric gross weights were selected for each aircraft design concept, and hover performance, installed power, fuel requirements, and aircraft subsystem weights were calculated for each parametric gross weight. Each aircraft resulted in a different payload. From the trends developed, the final design-point aircraft was determined which carried the 1200-pound design payload for the design mission. The parametric studies produced design-point characteristics for each of the five aircraft with five different single-engine propulsion systems. Mission suitability analyses were undertaken for each of the five aircraft.

Aircraft Drag Characteristics

A drag buildup was calculated for all aircraft, and a drag summary has been included for the single-engine aircraft powered by regenerative and nonregenerative engines. Drag was determined by considering the dimensions and shape of each aircraft component, and the drag estimation was based on Reference 11, in addition to data developed by Boeing.

Friction coefficients were calculated using a value of Reynolds number per unit length of 1.19×10^6 (which assumed a cruise speed of 145 kn at 4000 ft, 95°F ambient conditions). The results are summarized in Tables VI and VII.

Due to the basic similarity of the configurations with regenerative and nonregenerative engines, the total flatplate drag (F_e) was virtually the same.

Hover Performance

A 40-foot rotor diameter, which was the size of a growth BO-105 rotor, was selected because it was commensurate with the dimensions of the proposed helicopter. The 40-foot diameter also allowed a reasonable excursion in parametric disc loadings for the various gross weights considered.

The requirement that the aircraft be capable of withstanding 1.5 g's acceleration in hover without encountering rotorblade stall flutter established the rotor solidity. For blades of the type used in this study, the stall flutter boundary in terms of the ratio of thrust coefficient to solidity (C_T/σ) is shown in Figure 45. Solidity of the present BO-105 rotor, $\sigma = 0.07$, was selected as the minimum limiting value, to assure the structural and aeroelastic integrity of the rotor blade.

Tip speed, V_T = 716 ft/sec, was selected as representative of the growth BO-105 rotor. At the design cruise speed, then, the Mach number at the tip of the advancing blade would be less than 0.915, the Mach number at which compressibility effects would be encountered (established by airfoil thickness-chord ratio and section characteristics).

Figure 46, based upon nonuniform downwash velocities under a hovering rotor, was used to determine fuselage download in hover. An additional download was calculated as a function of disc loading and ambient conditions to provide 500 ft/min vertical rate of climb. A constant value of 1.5 percent of the download was estimated for trim and control.

Rotor figure-of-merit was plotted as a function of thrust coefficient and solidity in Figure 47. This relationship was obtained by analysis of a rotor having the dimensional and aerodynamic characteristics of the BO-105 rotor in the explicit vortex influence program developed by Boeing. The rotor system

TABLE VI. DRAG BUILDUP FOR AIRCRAFT WITH REGENERATIVE ENGINE (REYNOLDS NUMBER PER UNIT LENGTH = 1.19×10^6)

Component	Wetted Area, A_w (ft^2)	Increment in Flat-Plate Drag		Component F_e (ft^2)
		Percent	Incr. in F_e	
Fuselage	358.			
Basic F_e			1.096	
3-Dimensional Effects			.302	
Afterbody			1.485	
Canopy			.098	
Excrescences (Including Sliding Doors)			.431	3.412
Main Rotor Pylon/Engine Nacelle				
Basic F_e (Including Interference, 3-D Effects)			.668	
Excrescences		25.	.167	.835
Horizontal Tail	24.3			
Basic F_e			.0896	
3-Dimensional Effects		28.	.0251	
Interference			.0059	
Excrescences		4.	.0046	.125
Vertical Tail	18.65			
Basic F_e			.0671	
3-Dimensional Effects		160.	.1075	
Interference			.0861	
Excrescences		4.	.0070	.268
Nacelles				
3-Dimensional Effects				
Interference				
Excrescences				
Inlets & Exhausts				.278
Rotor Hubs				
Main Rotor Hubs (Including Hub-Pylon Interference)			3.000	
Tail Rotor			1.400	4.400
Miscellaneous				
Roughness (5% of $\Sigma C_F A_w$)			.125	
Cooling Momentum			.220	
Landing Skids			1.439	1.784
TOTAL				11.102

TABLE VII. DRAG BUILDUP FOR AIRCRAFT WITH SIMPLE-CYCLE ENGINE (REYNOLDS NUMBER PER UNIT LENGTH = 1.19×10^6)

Component	Wetted Area, A_w (ft^2)	Increment in Flat-Plate Drag		Component F_e (ft^2)
		Percent	Incr. in F_e	
Fuselage	358.			
Basic F_e			1.090	
3-Dimensional Effects			.300	
Afterbody			1.485	
Canopy			.098	
Excrescences (Including Sliding Doors)			.430	3.403
Main Rotor Pylon/Engine Nacelle				
Basic F_e (Including Interference, 3-D Effects)			.668	
Excrescences		25.	.167	.835
Horizontal Tail	24.3			
Basic F_e			.0896	
3-Dimensional Effects		28.	.0251	
Interference			.0059	
Excrescences		4.	.0046	.125
Vertical Tail	18.65			
Basic F_e			.0671	
3-Dimensional Effects		160.	.1075	
Interference			.0861	
Excrescences		4.	.0070	.268
Nacelles				
3-Dimensional Effects				
Interference				
Excrescences				
Inlets & Exhausts				.164
Rotor Hubs				
Main Rotor Hubs (Including Hub-Pylon Interference)			3.000	
Tail Rotor			1.400	4.400
Miscellaneous				
Roughness (5% of $\Sigma C_F A_w$)			.125	
Cooling Momentum			.220	
Landing Skids			1.439	1.784
Total				10.979

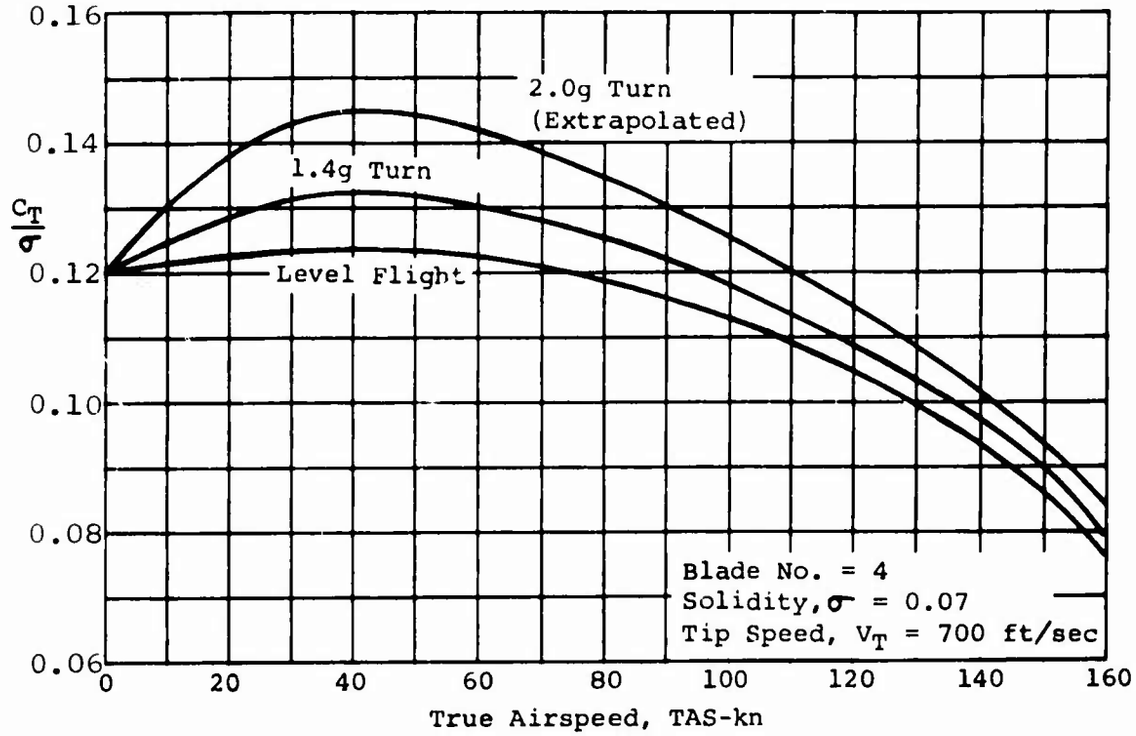


Figure 45. Estimated Moment-Stall Boundaries for the BO-105 Rotor.

design parameters and constraints were applied to obtain the installed power required for each configuration under the following takeoff conditions:

1. 4000 ft, 95°F ambient conditions

2. 500 ft/min vertical rate of climb

3. Takeoff at 95 percent of Military Rated Power

The ratio of rotor thrust to gross weight (including the above considerations of downloads, and rotor figure-of-merit were utilized in calculating rotor horsepower:

$$RHP = \frac{\left(\frac{T_R}{DGW}\right)^{3/2} DGW \sqrt{\frac{DGW}{A}}}{550\ FM \sqrt{2\rho}} \qquad (1)$$

Efficiency of the reduction gearbox was assumed to be 0.97.

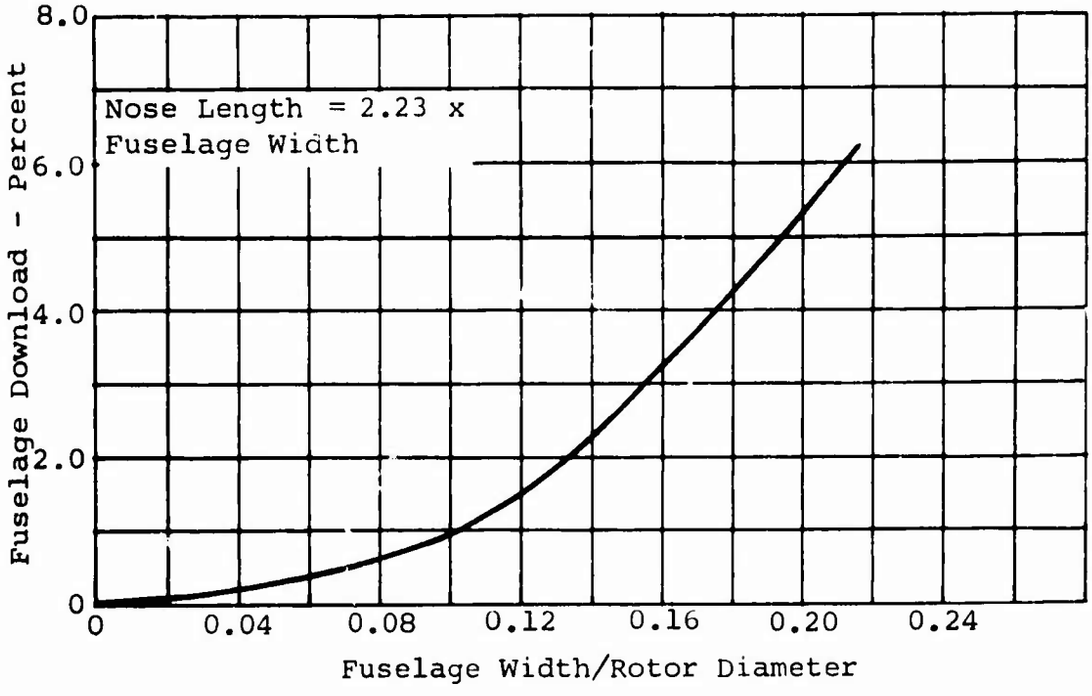


Figure 46. Single-Rotor Helicopter Fuselage Download.

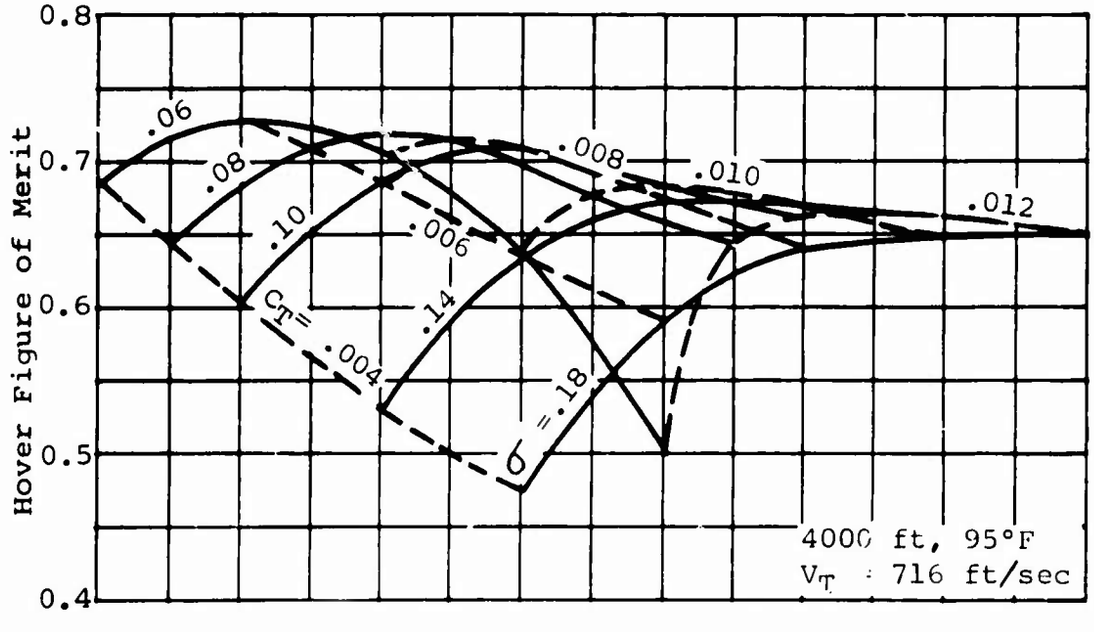


Figure 47. Carpet Plot of Hover Performance Parameters for BO-105 Type Rotor.

The tip speed of the two-bladed tail rotor was 716 ft/sec, and its solidity was 0.11, resulting in an efficiency of 0.712. These parameters were used in defining 95 percent Military Rated Power at the altitude, hot-day takeoff condition, and subsequently the installed power of the engine at sea level, 59°F.

Cruise Performance

Cruise power required for each configuration was determined from generalized rotor maps, developed from BO-105 helicopter data. Basic performance data for this aircraft are shown in Figure 48. Rotor horsepower was established from these basic curves of installed power required for various gross weights, eliminating transmission losses and tail rotor power. The data were generalized for ease of application in the parametric studies, in terms of the generalized rotor lift parameter, $\overline{L}$, and the generalized rotor power parameter, $\overline{P}$:

$$\overline{L} = \frac{GW}{qD^2\sigma} \qquad (2)$$

$$\overline{P} = \frac{325.8\ RHP}{qD^2\sigma\ V_{CR}} \qquad (3)$$

The generalized rotor performance map was plotted in Figure 49. An adjustment is required for the value of $\overline{P}$ obtained from Figure 49 (for the BO-105 helicopter) to account for the difference in F_e and D^2 applicable to the aircraft being studied, according to the following equations:

$$\overline{P}_{TOT} = \overline{P}\ (\text{Figure 49}) + \Delta\overline{P}_{REF} + \frac{\Delta F_e}{D^2\sigma} \qquad (4)$$

$$\Delta\overline{P}_{REF} = \frac{F_{e_{REF}}}{D^2\sigma} - \frac{F_{e_{REF}}}{(D^2\sigma)_{REF}} \qquad (5)$$

$$F_{e_{REF}} = 12.4\ ft^2$$

$$(D^2\sigma)_{REF} = 72.5\ ft^2$$

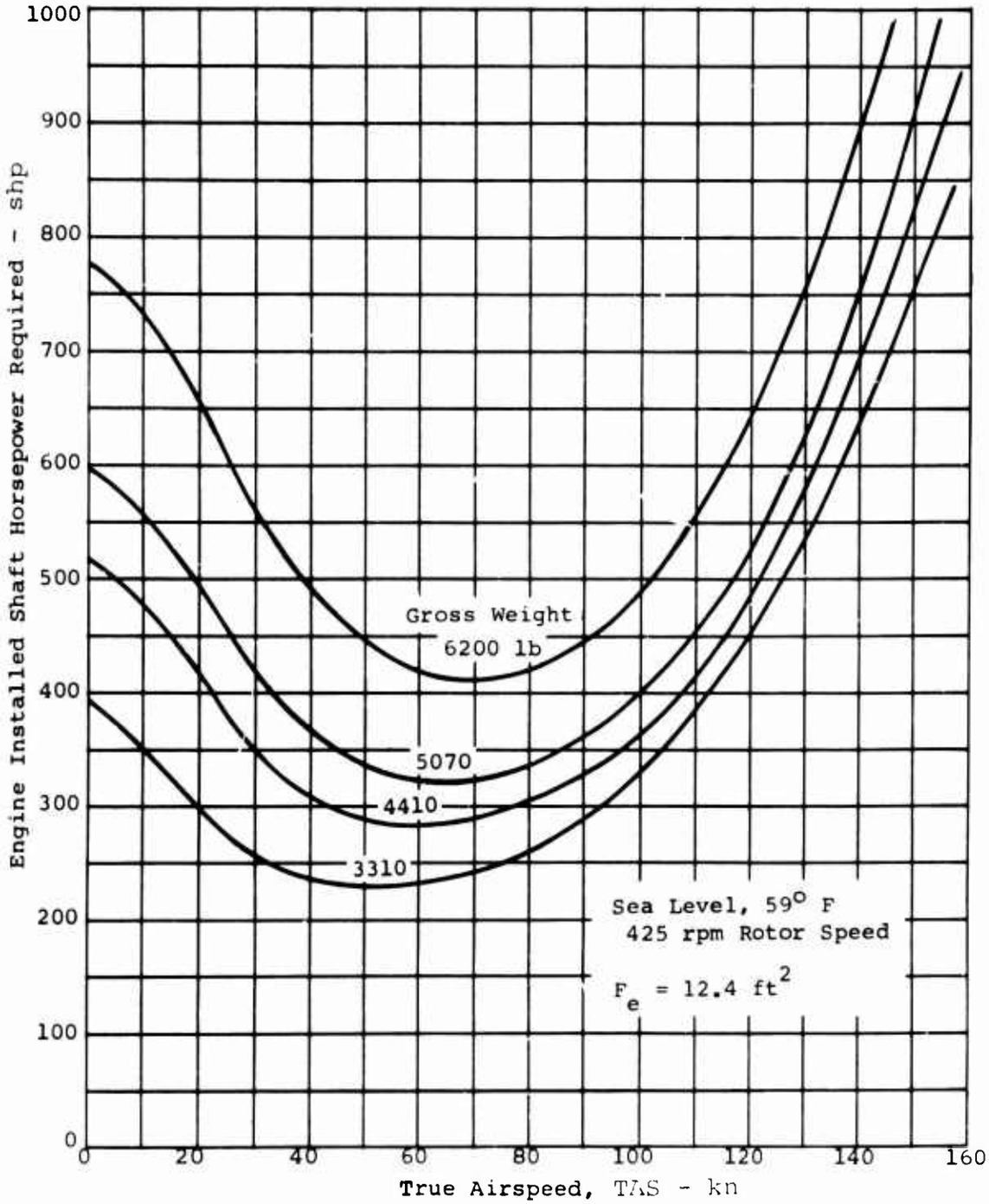


Figure 48. Installed Power Required for the BO-105 Helicopter.

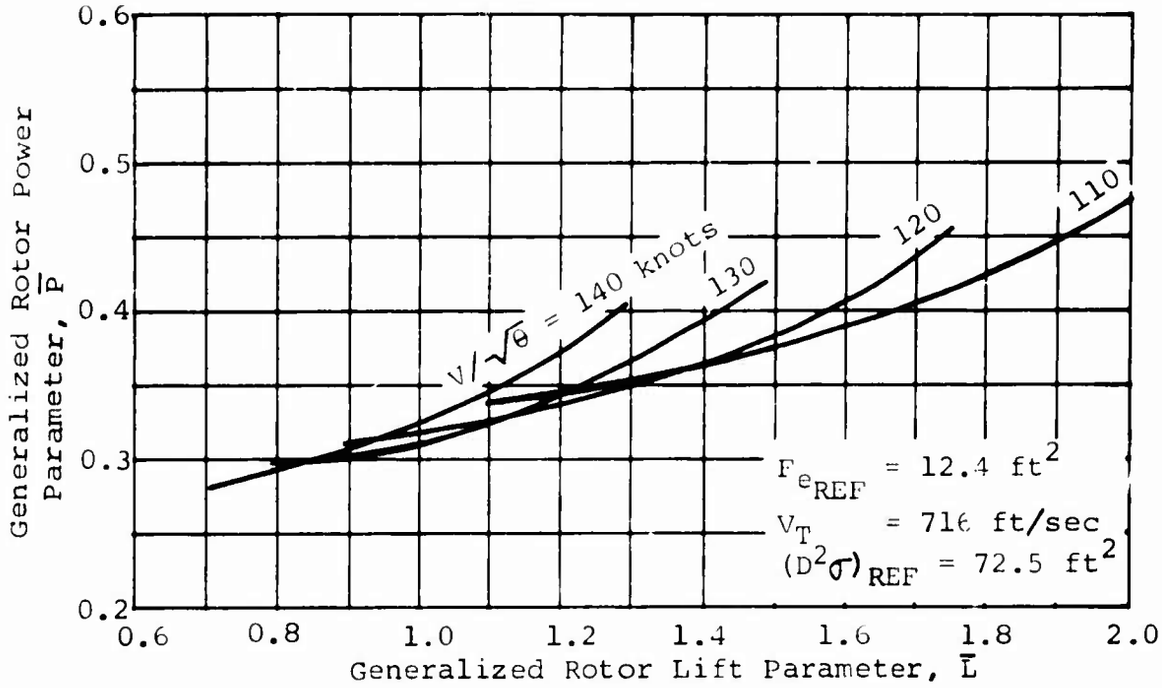


Figure 49. Generalized Rotor Performance Map for the BO-105 Rotor.

$$\Delta F_e = F_e - F_{e_{REF}} \tag{6}$$

$$SHP = \frac{\bar{P}_{TOT}\, q\, V_{CR}\, D^2\sigma}{325.8\,\eta_T} \tag{7}$$

The advance ratio and solidity for the rotors of the aircraft being studied were chosen to be consistent with the range of data for the BO-105, so no corrections in these parameters were necessary.

In the calculation of mission fuel, the aircraft weight was stepped for each segment of the mission. Therefore, cruise power was maintained constant for each segment, and the weight decrease as fuel was used was accounted for in increased cruise velocity, as follows:

$$\bar{V}_{CR} = \left[\frac{-.01V_{CR_o}^2 + 1.8V_{CR_o}}{2}\right]\left[\frac{\frac{\text{Length Segment}}{V_{CR_o}}\,(SFC)(SHP)}{GW_o}\right] + V_{CR_o} \tag{8}$$

Performance and weight data were calculated for each aircraft conceptual design at three disc loadings and three corresponding gross weights. The aircraft were designed to cruise at Normal Rated Power. Then, the payload capability of each parametric aircraft was determined as a function of gross weight and the final selected design-point helicopters were chosen to match the design payload of 1200 pounds.

Significant performance and weight parameters are summarized in Table VIII for the five design-point aircraft configurations.

TABLE VIII. SUMMARY OF SINGLE-ENGINE HELICOPTER CONFIGURATIONS PERFORMING DESIGN UTILITY MISSION

	Regenerative Engines			Non-regenerative Engines	
	Effectiveness			Adv.	Avail.
	0.40	0.65	0.80	Tech.	Tech.
Design Gross Weight, lb	6660	6525	6600	6620	7050
Weight Empty, lb	4211	4151	4257	4162	4494
Fixed Useful Load, lb (Incl. Mission Equipment)	780	780	780	780	780
Mission Fuel, lb	469	394	363	478	576
Payload, lb	1200	1200	1200	1200	1200
Disc Loading, lb/ft^2	5.30	5.18	5.25	5.27	5.62
Installed Power, shp (Sea level, 59°F, MRP)	1155	1120	1145	1165	1292
Transmission Rating, shp (4000 ft, 95°F, MRP)	855	832	846	850	925
Tail Rotor Power, shp	67	65	67	66	72
Total Equivalent Flat-plate Area, F_e, ft^2	11.10	11.10	11.10	10.98	10.98
V_{CR}, TAS, kn	143.5	143.0	143.2	141.7	145.8
V_{OPT}, TAS, kn	75.	75.	75.	77.	78.

Alternate Mission Suitability Analyses

Although the five aircraft were designed to perform a utility transport mission, the suitability of these helicopters for secondary medical-evacuation and observation missions was also investigated, and the results are summarized in Tables IX and X.

When equipped for the medical-evacuation mission, the helicopter required a small auxiliary fuel system which could be readily installed and fitted easily within the space of the unoccupied cargo area. The maximum gross weight for this mission exceeded design takeoff gross weight by approximately 1.0 percent, which was within the engine capability at Military Rated Power. However, internal fuel capacity was adequate to perform the observation mission, and the takeoff gross weight was less than the design gross weight.

Aircraft Weights

Weights for the various subsystems were developed from standard Boeing trend data. The following assumptions and ground rules were established.

1. Main and tail rotor diameters and tip speeds and fuselage configuration were considered constant.

2. The fixed equipment list was based on an austere approach, similar to the Light Observation Helicopters, OH-6A and OH-58A. Armament, armor, and specific mission equipment were included in useful load.

3. Since the utility and observation missions both required VFR conditions for observation, it was assumed that no all-weather avionics would be required. Navigational equipment would include only the simplest systems.

The weight trend equations used in the study are presented in Table XI. Those parameters which were constant throughout the analyses were itemized, together with a list of fixed equipment. A contingency allowance equal to 0.5 percent of design gross weight was included in the gross weight.

TABLE IX. SUMMARY OF SINGLE-ENGINE HELICOPTER CONFIGURATIONS PERFORMING MEDICAL EVACUATION MISSION

	Regenerative Engines			Nonregenerative Engines	
	Effectiveness			Adv. Tech.	Avail. Tech.
	0.40	0.65	0.80		
Weight Empty, lb	4211.	4151.	4257.	4162.	4494.
Fixed Useful Load, lb	620.	620.	620.	620.	620.
Mission Equipment, lb	270.	270.	270.	270.	270.
Medical Attendant, lb	200.	200.	200.	200.	200.
Litters (4, 15 lb each), lb	60.	60.	60.	60.	60.
Auxiliary Fuel System, lb	23.	21.	20.	22.	26.
Fuel (Main Tanks), lb	469.	394.	363.	478.	576.
Fuel (Auxiliary), lb	96.	83.	77.	89.	118.
Total Fuel, lb	565.	477.	440.	567.	694.
Takeoff Gross Weight, lb Warm Up, Take Off Cruise Out 75 NM	5949.	5799.	5867.	5901.	6364.
Landing Weight, lb Load 4 Litter Patients at 235 lb each	5776. +940.	5653. +940.	5733. +940.	5728. +940.	6152. +940.
Takeoff Gross Weight, lb* Take Off, Cruise 15 NM	6716.	6593.	6673.	6668.	7092.
Landing Weight, lb Unload 4 Litter Patients	6679. -940.	6561. -940.	6645. -940.	6630. -940.	7046. -940.
Takeoff Weight, lb Take Off, Cruise 15 NM	5739.	5621.	5705.	5690.	6106.
Landing Weight, lb Load 4 Litter Patients	5703. +940.	5591. +940.	5677. +940.	5653. +940.	6062. +940.
Takeoff Weight, lb Take Off, Cruise 15 NM	6643.	6531.	6617.	6593.	7002.
Landing Weight, lb Unload 4 Litter Patients	6606. -940.	6499. -940.	6589. -940.	6555. -940.	6956. -940.
Takeoff Weight, lb Take Off, Cruise 15 NM	5666.	5559.	5649.	5615.	6016.
Landing Weight, lb Load 4 Litter Patients	5630. +940.	5529. +940.	5621. +940.	5578. +940.	5972. +940.
Takeoff Weight, lb Take Off, Cruise 15 NM	6570.	6469.	6561.	6518.	6912.
Landing Weight, lb Unload 4 Litter Patients	6533. -940.	6437. -940.	6533. -940.	6480. -940.	6866. -940.
Takeoff Weight, lb Take Off Cruise Return 75 NM	5595.	5497.	5593.	5540.	5926.
Landing Weight, lb at Base	5441.	5370.	5471.	5391.	5739.

* The minimum gross weight for takeoff during this mission exceeded the design takeoff gross weight by approximately 1.0%. This was within the hover capability at MRP.

TABLE X. SUMMARY OF SINGLE-ENGINE HELICOPTER CONFIGURATIONS PERFORMING OBSERVATION MISSION

	Regenerative Engines			Nonregenerative Engines	
	Effectiveness			Adv. Tech.	Avail. Tech.
	0.40	0.65	0.85		
Weight Empty, lb	4211.	4151.	4257.	4162.	4494.
Fixed Useful Load, lb	620.	620.	620.	620.	620.
Mission Equipment, lb	120.	120.	120.	120.	120.
Observer, lb	200.	200.	200.	200.	200.
Fuel (Main Tanks), lb*	469.	394.	363	478.	576.
Takeoff Gross Weight, lb Warm Up, Take Off Cruise Outbound 10 NM	5620.	5485.	5560.	5580.	6010.
Aircraft Weight, lb (Start of Observation) V = 82.5 kn - TAS Time on Station = 2 hr	5583.	5455.	5532.	5543.	5965.
Aircraft Weight, lb (Start Return to Base) Cruise Return 10 NM	5233.	5147.	5246.	5177.	5523.
Landing Weight, lb	5210.	5130.	5230.	5154.	5495.

*Internal fuel was adequate to perform observation mission.

TABLE XI. WEIGHT TREND EQUATIONS AND CONSTANT PARAMETERS FOR AIRCRAFT PARAMETRIC CONFIGURATION STUDIES

Group	Weight Trend Equations	Constant Parameters
Rotor Group	$W_R = J_R \left[r^{0.25} \left(\frac{RHP}{100}\right)^{0.5} \left(\frac{V_{T_L}}{100}\right) \left(\frac{R b_R c_R}{10}\right) K_D \right]^{0.67}$	J_R = 17.05 r = .0756R = 1.512 ft V_{T_L} = 1.07 V_T = 766 ft/sec R = 20. ft b = 4 K_D = 1.0
Tail Group		
Tail Rotor	$W_{TR} = 14.65 \left[r_{TR}^{0.25} \left(\frac{HP_{TR}}{100}\right)^{0.5} \left(\frac{V_{T_L}}{100}\right) \left(\frac{R_{TR} b_{TR} c_{TR}}{10}\right) \right]^{0.67}$	r_{TR} = 0.182 R_{TR} V_{TL} = 1.07 V_T = 766 ft/sec b_{TR} = 2
Horizontal Tail	$W_{HT} = 10.0 \left[\left(\frac{DGW}{10^4}\right) \eta_U \left(\frac{S_H}{10}\right) \left(\frac{TMA_H}{10}\right) \left(\log_{10} V_{MAX} \right) \right]^{0.375}$	η_U = 5.25 S_H = 14.8 ft^2 S_V = 9.2 ft^2
Vertical Tail	$W_{VT} = 3.55 \left[\left(\frac{DGW}{10^4}\right) \eta_U \left(\frac{S_V}{10}\right) \left(\frac{TMA_V}{10}\right) (1+h) \left(\log_{10} V_{MAX} \right) \right]^{0.575}$	TMA_H = 18.15 ft TMA_V = 22.75 ft V_{MAX} = 160 kn h = 3.125 ft
Body Group	$W_{BG} = 125 \left\{ \left[\left(\frac{DGW}{10^4}\right) \eta_U \left(\frac{S_f}{10^3}\right) \left(L_c + L_{rw} + \Delta_{CG}\right) \right]^{0.5} \log_{10} V_{MAX} \right\}^{0.8} + 0.05 W_R$	η_U = 5.25 V_{MAX} = 160 kn L_C = 17.9 ft Δ_{CG} = 1.0 ft
Landing Gear Group	$W_{LG} = 0.022$ DGW	
Flight Controls Group	$W_{FC} = 0.455 W_R + 25.5 \left(\frac{W_R}{100}\right)^{0.84} + 45.$	
Engine Section		
Engine Mounts	$W_{MT} = N_e \left(W_e \eta_{cr} \right)^{0.41}$	η_{cr} = 20. for Internal Mounting η_{cr} = 12. for External Mounting
Cowling	Cowling structure weights were calculated from layouts for the various engine installations.	
Propulsion Group		
Engine Dry Weight	$W_e = K_1 (SHP^*) + K_2$ Weights for engine subsystems, including the air induction system and particle separator, exhaust system, cooling system, lubrication system, controls, and starting system, were estimated or calculated based on engine installation layouts. Starting system weight was included in the electrical group weight.	Engine — K_1 lb/shp — K_2 lb Regenerative 0.60 Effectiveness — .12 — 35. 0.65 Effectiveness — .1447 — 36. 0.80 Effectiveness — .2103 — 44. Simple-Cycle Advanced-Technology — .0933 — 33. Available-Technology — .20 — 38.
Fuel System	$W_{FS} = 0.681 W_F + 145.$	
Drive System	$W_D = 271 \left(\frac{HP_X}{N_R}\right)^{0.8}$	
Fixed Equipment	Auxiliary Powerplant Group — 0 lb Instrument Group — 23 Hydraulics and Pneumatics Group — 0 Electrical Group — 125 Electronics Group — 158 Armament Group — 50 Furnishings and Equipment Group — 60 Air-Conditioning and De-icing — 30 Total — 446 lb The fixed equipment list was based on the assumptions and ground rules established for the weights analyses.	
Vibration Control Devices	$W_{VCD} = 0.01 \left[\frac{DGW \left(V_{MAX} - 50\right)^3}{(N_R b)^{1.5} \times b} \right] + 30.$	

The fixed useful load, mission equipment, and payload for the design mission and secondary missions are itemized below. The fixed useful load was common for all missions, while mission equipment and payload were different for the secondary missions.

1.	Fixed Useful Load:		620 lb
	Crew (3 at 200 lb each)	600 lb	
	Trapped Liquids	10	
	Engine Oil	10	
2.	Design Mission - Utility:		
	Fixed Useful Load		620 lb
	Mission Equipment		160 lb
	Troop Seats	25	
	5.56 MM Machine Gun	12	
	5.56 MM Ammunition	100	
	Communications Package	23	
	Required Payload		1200 lb
	Total		1980 lb
3.	Medical-Evacuation Mission:		
	Fixed Useful Load		620 lb
	Mission Equipment		270 lb
	Litter Supports and Stanchions	48	
	Medic's Seat	7	
	Medical Equipment	100	
	Machine Gun, Ammunition	112	
	Miscellaneous Equipment	3	
	Required Payload		1200 lb
	Total		2090 lb
4.	Observation Mission		
	Fixed Useful Load		620 lb
	Mission Equipment		120 lb
	Observer's Seat	7	
	Machine Gun, Ammunition	112	
	Miscellaneous	1	
	Required Payload		200 lb
	Total		940 lb

PROPULSION/AIRFRAME INTEGRATION

As discussed in the AIRCRAFT CONFIGURATION STUDIES section of this report, conceptual designs of single-engine utility helicopters retained for this and subsequent tasks included:

1. A simple-cycle advanced-technology engine
2. A simple-cycle available-technology engine
3. Regenerative advanced-technology engines with regenerators having three different values of effectiveness

Engine installations were integrated with the airframe, optimized with respect to inlet, exhaust, subsystems, orientation, and other aspects to provide the best possible aircraft weight, balance, and drag characteristics. Detailed weight analyses were performed and vehicle summary weight statements were prepared and tabulated in Appendix III. Final three-view aircraft drawings and propulsion system installation drawings were prepared for the helicopters with regenerative engines having 0.40 and 0.80 effectiveness and for the helicopter powered by the simple-cycle available-technology engine, and these drawings have been included in the text. Use of the advanced-technology simple-cycle engine would produce a design identical to the available-technology engine, except for physical dimensions of the engine, and would ease installation problems.

PROPULSION SYSTEM INTEGRATION DESIGN

One-quarter scale propulsion system installation drawings were prepared for each aircraft to identify detail problems. The drawings for the two regenerative engines and one simple-cycle engine are discussed in the following paragraphs.

Regenerative Engines

The 0.40 effectiveness regenerative engine is pictured in Figure 50 - a rear-drive engine with 40,000 rpm output shaft speed, scaled from the B-2 configuration presented in Reference 1 to the required 1155-shp. The engine was mounted forward of the main transmission. The engine intake is at the forward end of the cowl. The radial flow of hot gas at the exit of the recuperator is collected and discharged through elbows which exhaust to port and starboard, aft and beyond the cowl mold lines.

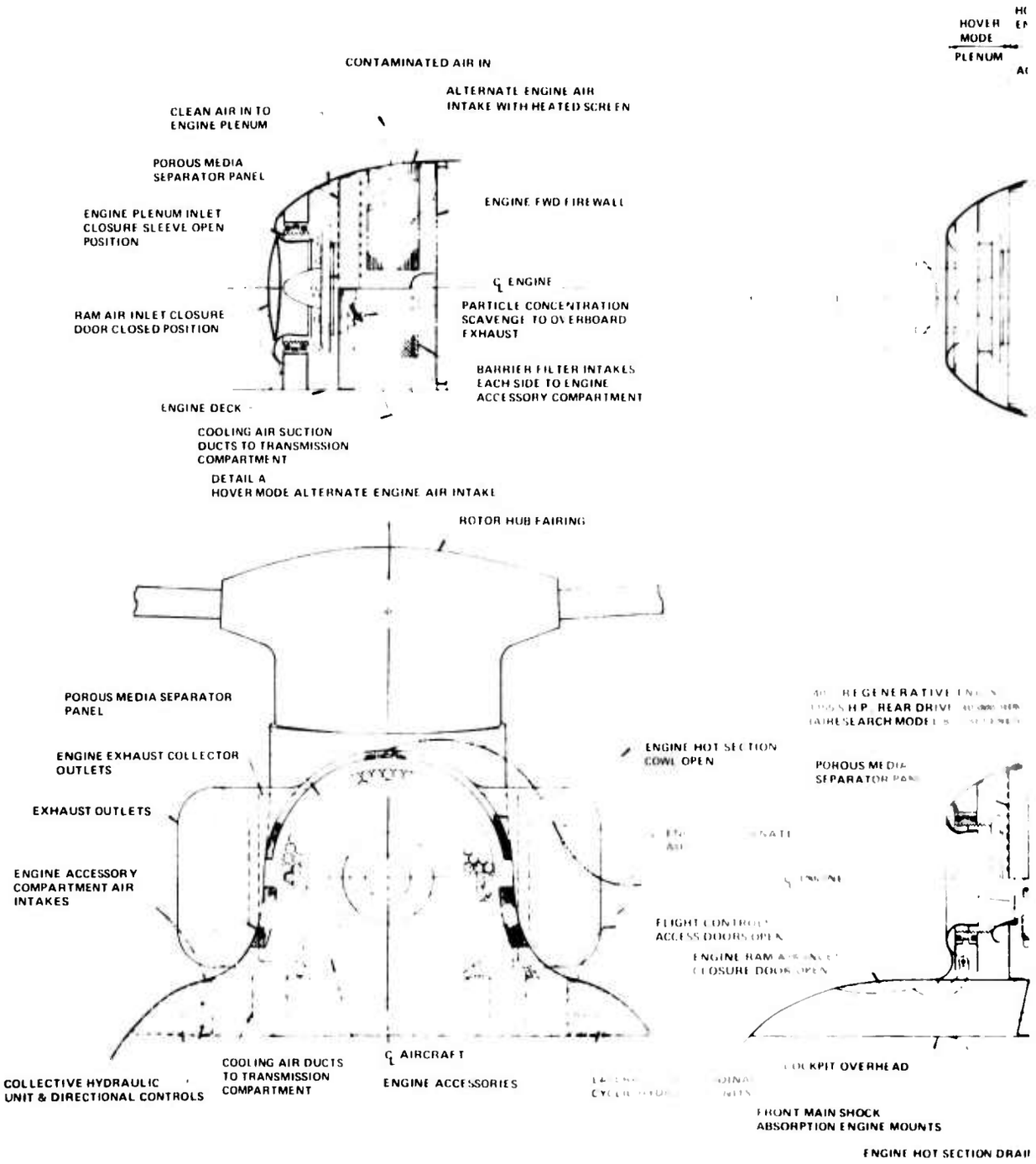
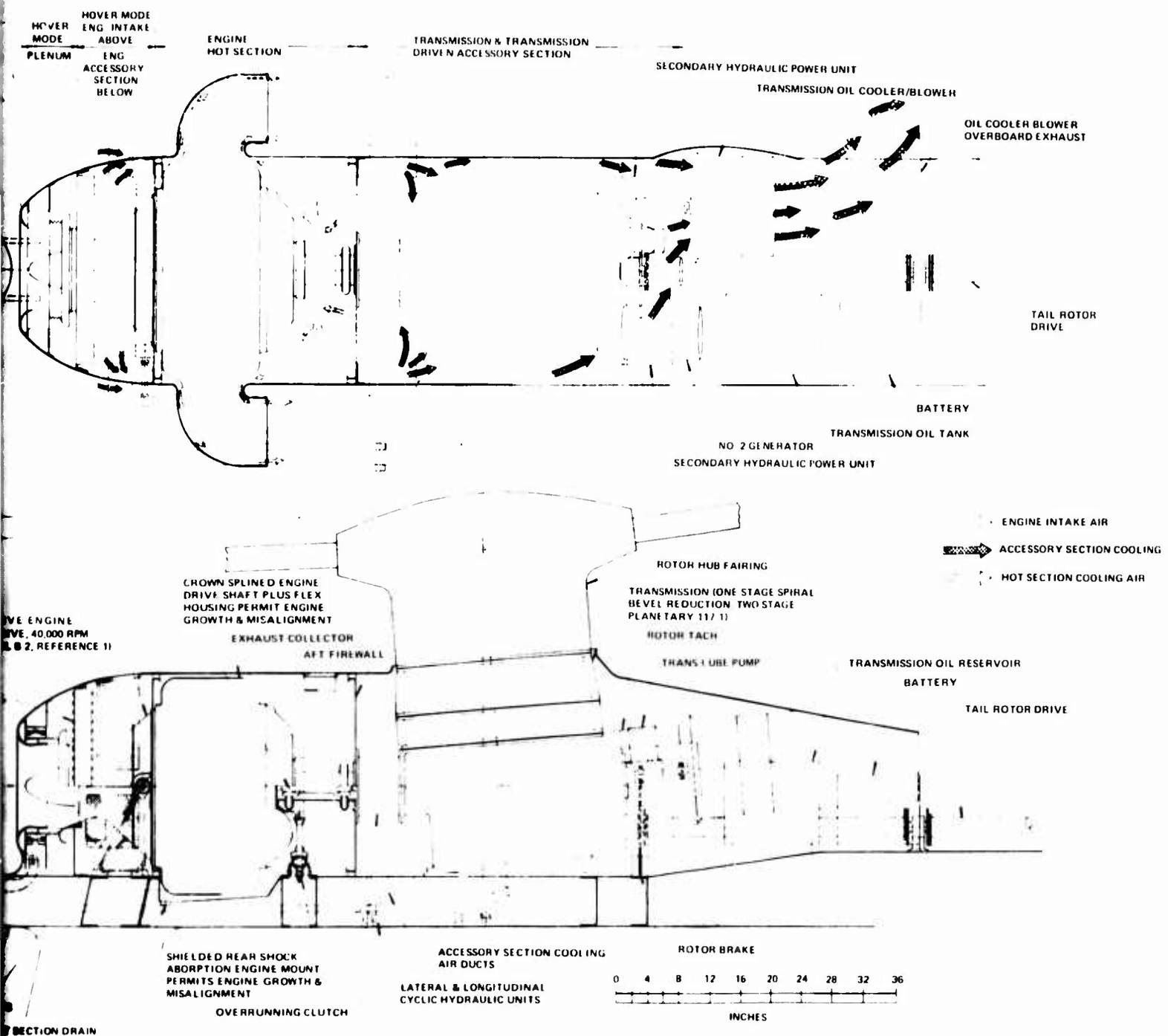


Figure 50. Propulsion System Installation, Single Advanced-Technology Regenerative Engine (0.40 Effectiveness) in Utility Helicopter.


HOVER MODE ENG INTAKE ABOVE
HOVER MODE PLENUM
ENG ACCESSORY SECTION BELOW
ENGINE HOT SECTION
TRANSMISSION & TRANSMISSION DRIVEN ACCESSORY SECTION
SECONDARY HYDRAULIC POWER UNIT
TRANSMISSION OIL COOLER/BLOWER
OIL COOLER BLOWER OVERBOARD EXHAUST
TAIL ROTOR DRIVE
BATTERY
TRANSMISSION OIL TANK
NO 2 GENERATOR
SECONDARY HYDRAULIC POWER UNIT
ENGINE INTAKE AIR
ACCESSORY SECTION COOLING
HOT SECTION COOLING AIR
ROTOR HUB FAIRING
CROWN SPLINED ENGINE DRIVE SHAFT PLUS FLEX HOUSING PERMIT ENGINE GROWTH & MISALIGNMENT
TRANSMISSION (ONE STAGE SPIRAL BEVEL REDUCTION TWO STAGE PLANETARY 11/1)
EXHAUST COLLECTOR
ROTOR TACH
AFT FIREWALL
TRANSMISSION OIL RESERVOIR
BATTERY
TAIL ROTOR DRIVE
SHIELDED REAR SHOCK ABORPTION ENGINE MOUNT PERMITS ENGINE GROWTH & MISALIGNMENT
OVERRUNNING CLUTCH
ACCESSORY SECTION COOLING AIR DUCTS
LATERAL & LONGITUDINAL CYCLIC HYDRAULIC UNITS
ROTOR BRAKE
0 4 8 12 16 20 24 28 32 36
INCHES

Power transfer to the main transmission is accomplished with a crown-splined coupling shaft, which would permit small amounts of slip and misalignment due to thermal growth of the engine, flight deflection, and manufacturing tolerances. The shaft is enclosed in a flexible housing and is lubricated from the main transmission.

Three-point shock-absorption engine mounts are provided. Two main mounts are located forward in the accessory section, and one center mount is located aft under the output drive shaft. The aft mount in the hot section allows engine growth, and could be shielded and cooled from below if required.

Firewalls are located fore and aft of the engine hot section. The forward firewall is attached to the engine flange between the accessory section and the exhaust collector. The engine deck and cowl are sealed against the firewall. The aft firewall shields the main transmission from the engine hot section, and the engine cowl and transmission access panels seal against it.

Engine hot-section cooling is accomplished by inducing a flow of ambient air through this compartment, using the engine exhaust to provide ejector action at the outlets of the collector elbows. Cooling air enters the compartment through a screened opening at the upper-aft end and is drawn forward and down over the engine as indicated by the cross-hatched arrows in the plan view of Figure 50. The engine accessory section is cooled by a flow of air through screened openings in the cowl on the port and starboard sides, immediately under the engine centerline. Cooling air is drawn over the accessory section by the oil cooling/blower, which is located in the main transmission section. The transmission oil cooler was sized to produce a constant suction in the accessory section cooling air ducts, as well as to provide a cooling flow over the transmission and transmission-driven accessories. Air from the oil cooler/blower is discharged from the starboard side of the cowl. Engine-accessory, transmission, and transmission-accessory cooling-air flow is indicated by checked/shaded arrows in the plan view.

Engine intake air is provided by a ram-air duct during forward flight by an alternate filtered-air source for hover operation. The hover system includes a screened intake and plenum and a particle separator panel to remove contaminants. A closure door for the ram air inlet during hover operation is included.

Detail "A" illustrates operation of the alternate air intake system.

The upper airframe structure was designed to provide a solid attachment for the main transmission as well as support for the engine, cowl, accessories, and drives. All components were mounted on top of the structure to be accessible and removable from the sides and top after opening or removing cowl segments.

The main engine mount supports can be unbolted at the deck and the rear mount at its attachment to the engine, resulting in quick engine removal. Engine weight is 174 pounds, and hoisting can be accomplished easily with a makeshift frame and a small block and tackle.

The main rotor hub, or the hub and transmission, can be removed as a unit. The transmission is attached to the structure with a bolted flange at its base. Flight controls can be disconnected for transmission removal, using an access panel from the cargo compartment.

The arrangement of dynamic components in this configuration enhanced overall balance of the aircraft.

The 0.80 effectiveness regenerative engine, pictured in Figure 51, was installed in an identical manner. This engine was only 2.25 inches larger in diameter (with exhaust collector in place) and 4.5 inches longer. The engine was scaled from the data in Reference 1 to the required 1145 shp, and it weighed 285 pounds.

Simple-Cycle Engine

Figure 52 presents the installation of an available-technology simple-cycle engine in the baseline airframe. A rear-drive configuration was used, to be comparative with the regenerative engine installations, with a 40,000-rpm output-shaft speed. The engine was sized for 1292 shp, resulting in an engine weight of 296 pounds. The front intake is conventional, and the exhaust hood concept is typical of rear-drive shaft engines. The engine diameter is 14.5 inches maximum, and its overall length is 36 inches. A simple ram intake duct is fitted to the front flange of the engine, and an exhaust elbow and tailpipe are used to turn the exhaust gas outboard on the port side of the cowl.

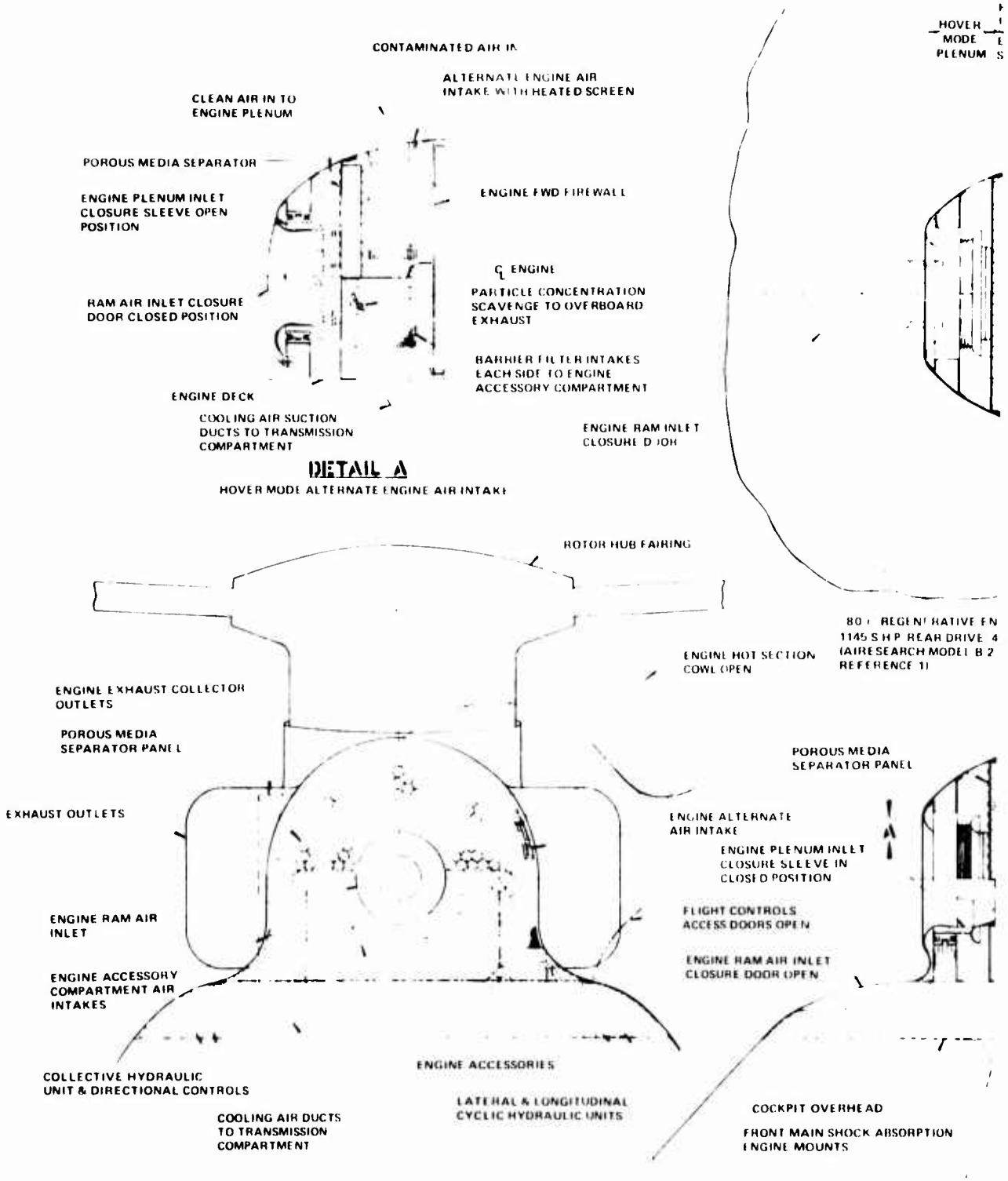
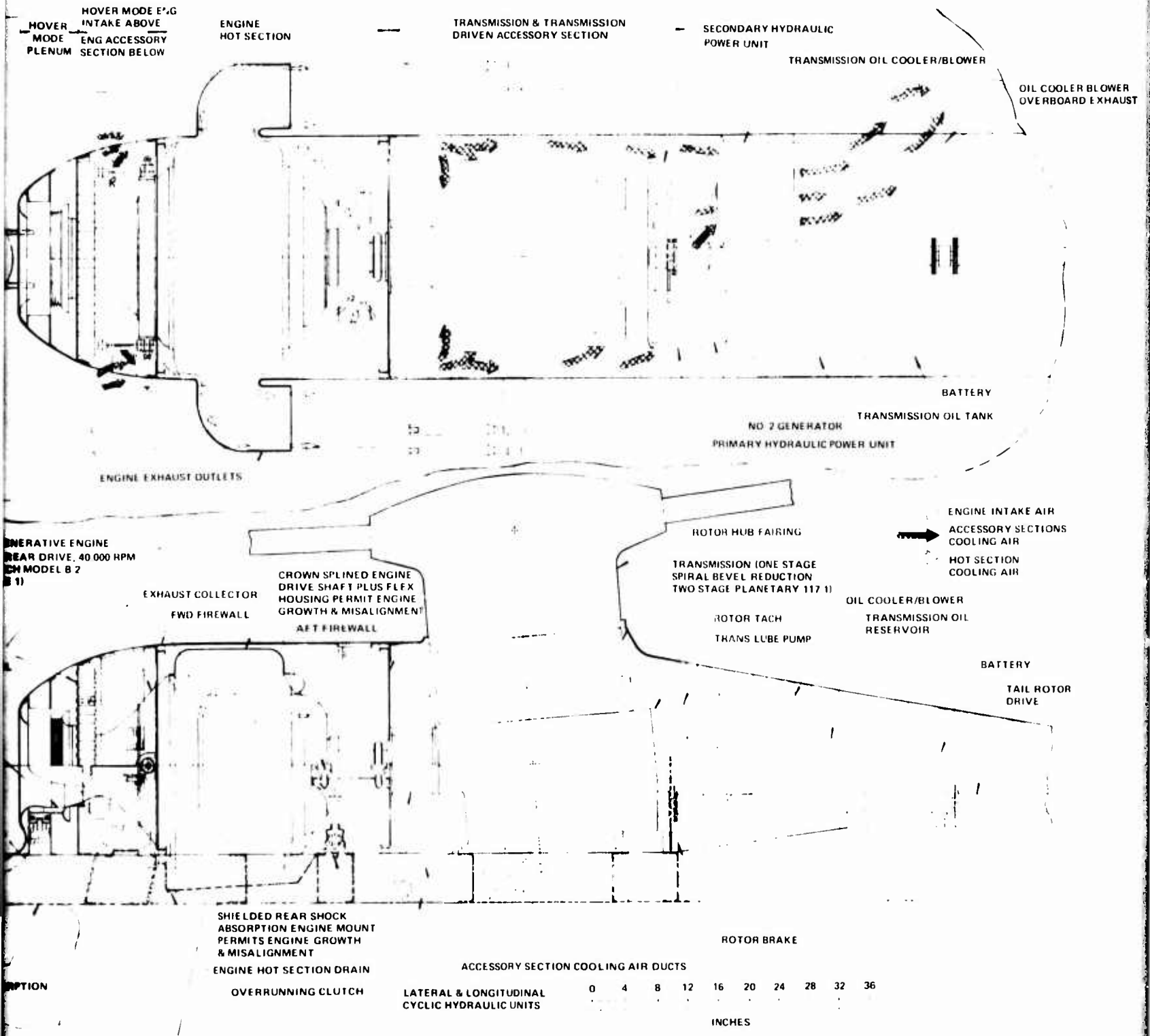


Figure 51. Propulsion System Installation, Single Advanced-Technology Regenerative Engine (0.80 Effectiveness) in Utility Helicopter.


HOVER MODE PLENUM
HOVER MODE ENG INTAKE ABOVE ENG ACCESSORY SECTION BELOW
ENGINE HOT SECTION
TRANSMISSION & TRANSMISSION DRIVEN ACCESSORY SECTION
SECONDARY HYDRAULIC POWER UNIT
TRANSMISSION OIL COOLER/BLOWER
OIL COOLER BLOWER OVERBOARD EXHAUST
BATTERY
TRANSMISSION OIL TANK
NO 2 GENERATOR
PRIMARY HYDRAULIC POWER UNIT
ENGINE EXHAUST OUTLETS
ENGINE INTAKE AIR
ACCESSORY SECTIONS COOLING AIR
HOT SECTION COOLING AIR
ROTOR HUB FAIRING
TRANSMISSION (ONE STAGE SPIRAL BEVEL REDUCTION TWO STAGE PLANETARY 117 1)
OIL COOLER/BLOWER
TRANSMISSION OIL RESERVOIR
ROTOR TACH
TRANS LUBE PUMP
BATTERY
TAIL ROTOR DRIVE
NERATIVE ENGINE
EAR DRIVE, 40 000 RPM
H MODEL B 2
1)
EXHAUST COLLECTOR
FWD FIREWALL
CROWN SPLINED ENGINE DRIVE SHAFT PLUS FLEX HOUSING PERMIT ENGINE GROWTH & MISALIGNMENT
AFT FIREWALL
SHIELDED REAR SHOCK ABSORPTION ENGINE MOUNT PERMITS ENGINE GROWTH & MISALIGNMENT
ENGINE HOT SECTION DRAIN
OVERRUNNING CLUTCH
ROTOR BRAKE
ACCESSORY SECTION COOLING AIR DUCTS
LATERAL & LONGITUDINAL CYCLIC HYDRAULIC UNITS
0 4 8 12 16 20 24 28 32 36
INCHES

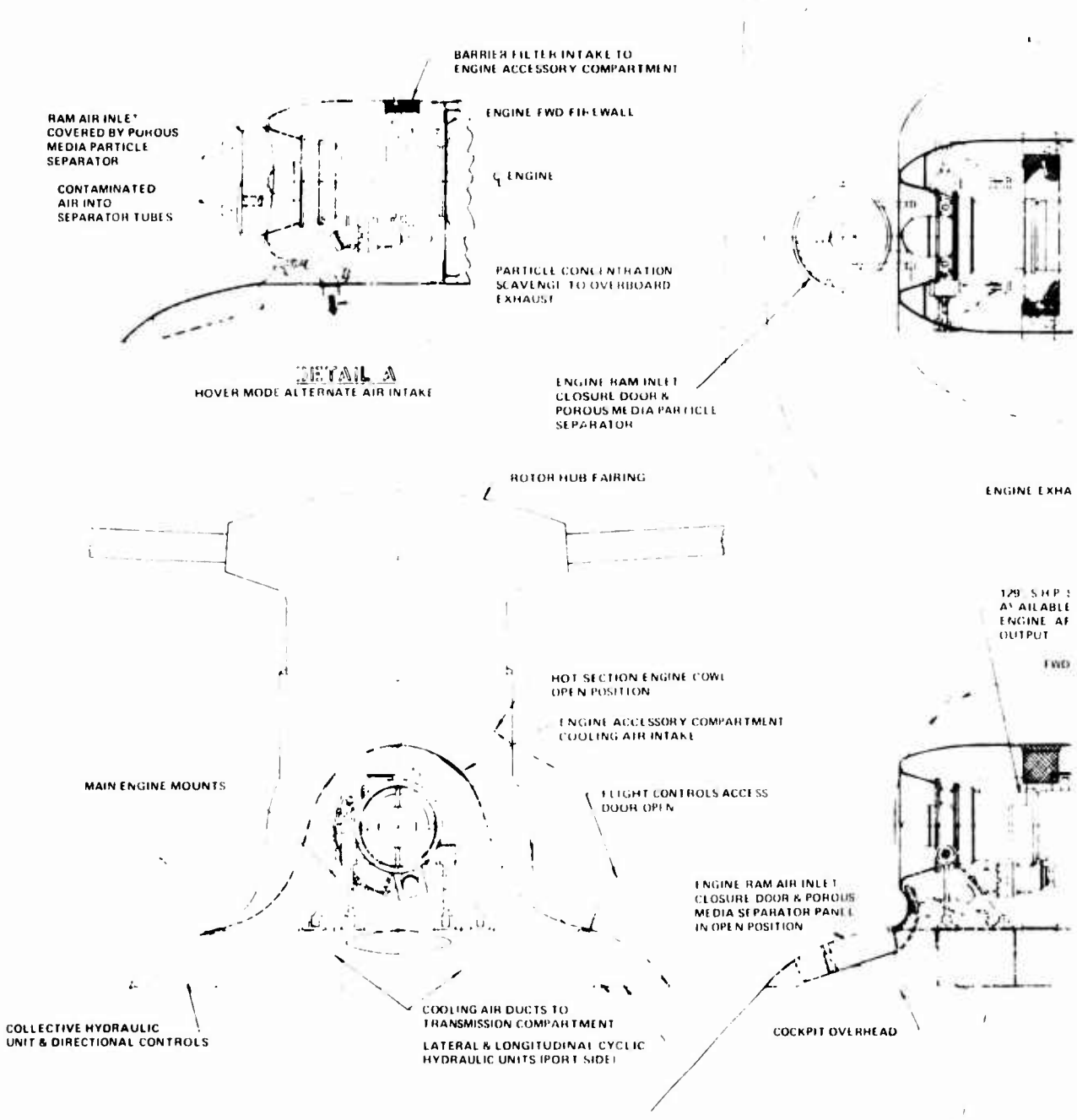
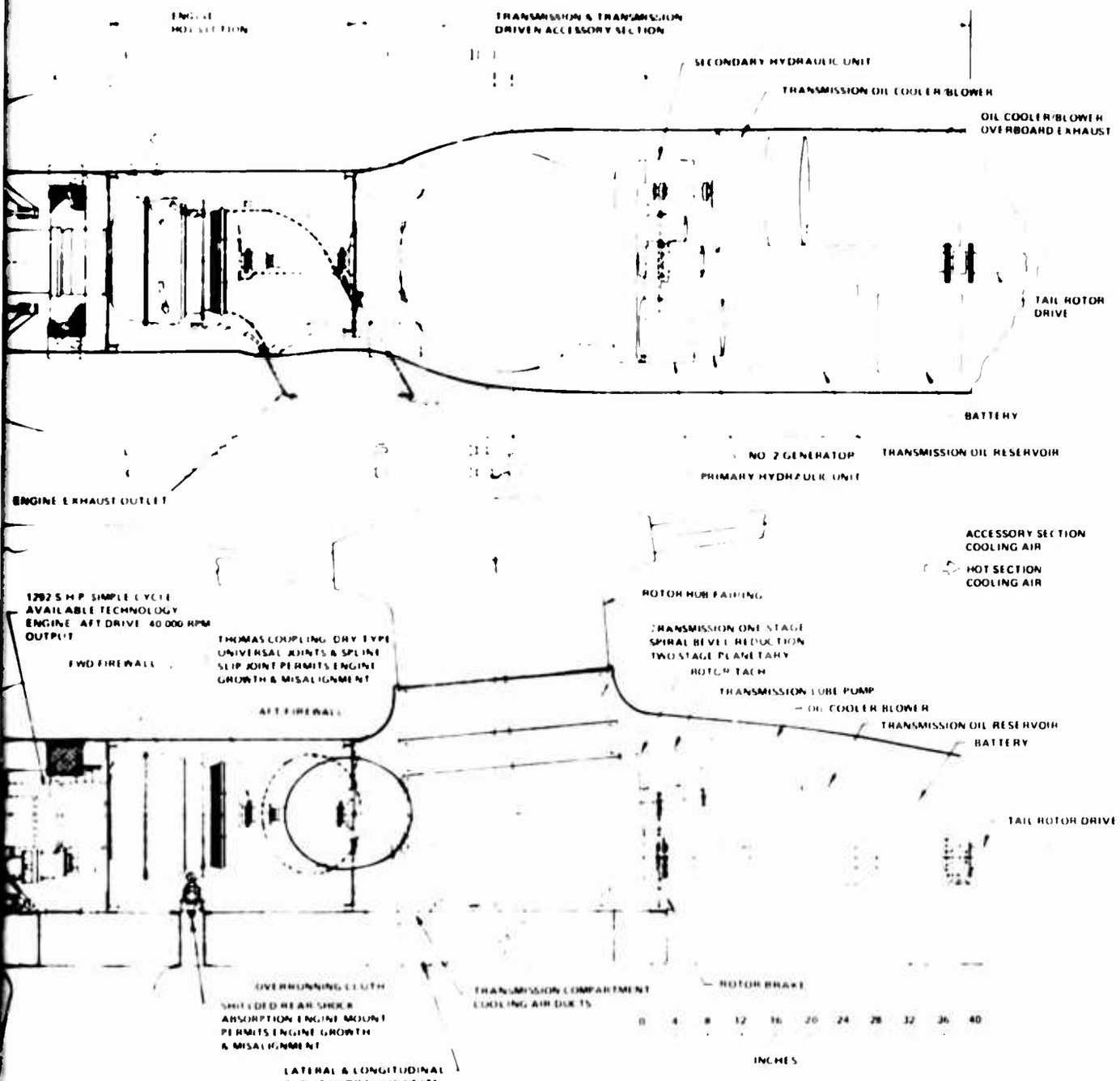


Figure 52. Proulsion System Installation, Single Available-Technology Simple-Cycle Engine in Utility Helicopter.


ENGINE
HOT SECTION
TRANSMISSION & TRANSMISSION
DRIVEN ACCESSORY SECTION
SECONDARY HYDRAULIC UNIT
TRANSMISSION OIL COOLER/BLOWER
OIL COOLER/BLOWER
OVERBOARD EXHAUST
TAIL ROTOR
DRIVE
BATTERY
NO 2 GENERATOR
TRANSMISSION OIL RESERVOIR
PRIMARY HYDRAULIC UNIT
ENGINE EXHAUST OUTLET
ACCESSORY SECTION
COOLING AIR
HOT SECTION
COOLING AIR
1292 S H P SIMPLE CYCLE
AVAILABLE TECHNOLOGY
ENGINE AFT DRIVE 40 000 RPM
OUTPUT
FWD FIREWALL
THOMAS COUPLING DRY TYPE
UNIVERSAL JOINTS & SPLINE
SLIP JOINT PERMITS ENGINE
GROWTH & MISALIGNMENT
AFT FIREWALL
ROTOR HUB FAIRING
TRANSMISSION ONE STAGE
SPIRAL BEVEL REDUCTION
TWO STAGE PLANETARY
ROTOR TACH
TRANSMISSION TUBE PUMP
OIL COOLER BLOWER
TRANSMISSION OIL RESERVOIR
BATTERY
TAIL ROTOR DRIVE
OVERRUNNING CLUTCH
SHIELDED REAR SHOCK
ABSORPTION ENGINE MOUNT
PERMITS ENGINE GROWTH
& MISALIGNMENT
TRANSMISSION COMPARTMENT
COOLING AIR DUCTS
ROTOR BRAKE
0 4 8 12 16 20 24 28 32 36 40
INCHES
LATERAL & LONGITUDINAL
CYCLIC HYDRAULIC UNITS

Power transfer to the main transmission is through a shaft approximately 12 inches long, using dry flexible couplings for misalignment and a slip joint.

As with the regenerative engine installations, a three-point shock absorption mount system is used. The two main mounts are located on the front frame, and the third is located in the hot section near the exhaust flange. This mount can be shielded and cooled from below if required.

Firewalls are located to isolate the engine hot section from the accessories and main transmission. The forward firewall is attached to the engine and sealed against the engine deck and cowl. The aft firewall is fitted against the overrunning clutch housing and shields the main transmission.

Cooling in the hot section and in the accessory section is accomplished in the manner shown previously for the regenerative engines. Cooling-air flow through the engine compartments is shown in the plan view in Figure 52.

A separator panel was incorporated into the ram air inlet closure door. During hover operation, this door can be raised to cover the engine inlet, interposing a particle separator panel in the path of the inflowing air. The separator panel is scavenged through a flexible duct ahead of a blower mounted in the fuselage. Detail "A" of Figure 52 illustrates engine airflow during hover operation.

VEHICLE SUMMARY WEIGHT STATEMENTS

Vehicle summary weight statements were prepared for all the single-engine aircraft conceptual designs. Detailed weight analyses were performed for all five design-point helicopters. The weight statements were prepared in the format described in MIL-STD-451, Part I. The results are presented in tabular form in Appendix III.

CONFIGURATION DESIGNS

Three-view general arrangement drawings of the utility helicopter designs powered by regenerative engines with 0.40 and 0.80 effectiveness recuperators, and the drawing of the aircraft with a simple-cycle engine are shown in Figures 53, 54, and 55.

The aircraft were designed using the baseline Army utility airframe configuration described earlier. In the regenerative engine installations, the exhaust gas was carried outboard and aft by means of a collector around the periphery of the engine and outlet elbows on the port and starboard sides. The simple-cycle engine used an exhaust elbow and tailpipe to turn the exhaust outboard on the port side of the cowl.

All dynamic components, including engine transmissions, drives, and rotors, were mounted on top of the airframe structure, providing good accessibility and protection from ground fire. Hinged and removable nonstructural cowls and fairings covered the engine, transmission, and drives.

The center portion of the main rotor transmission was identical in each of the aircraft configurations. The 40,000-rpm output-shaft speed for the engines was reduced to 342 rpm, the main rotor speed, through a one-stage spiral bevel (5.5:1) and a two-stage planetary (4.6:1 each) reduction gear.

Cyclic and collective flight controls in each configuration were routed aft from their respective sticks, then vertically up the center column between the cockpit and cargo compartment. Once overhead, torque tubes transmit control movements laterally to the upper shoulders of the fuselage. From these points, longitudinal push/pull rods transmit control motions aft to the center of the main rotor transmission. At this juncture a second set of torque tubes carries the control inboard to a point directly under the center of the main transmission. There, vertical rods inside the main rotor shaft, actuated by bell cranks under the transmission and through a swashplate above the rotor hub, impart control forces to the blades.

The fuel system comprises two separate self-sealing, belly-mounted tanks, each incorporating individual pumping, plumbing, and jettison provisions. A baffle and jet pump is included in the main cell to insure against starving the pump in hover operation. One-point pressure fueling is included for both tanks.

Basically, the airframe structure is a box surrounding the cargo compartment, with openings on two sides and in the rear. The propulsion system is located on top of the box, with the main rotor axis located directly over the center of gravity of the cargo compartment. A conical cross-section tail boom is gusseted to the aft side of the box at a convenient height,

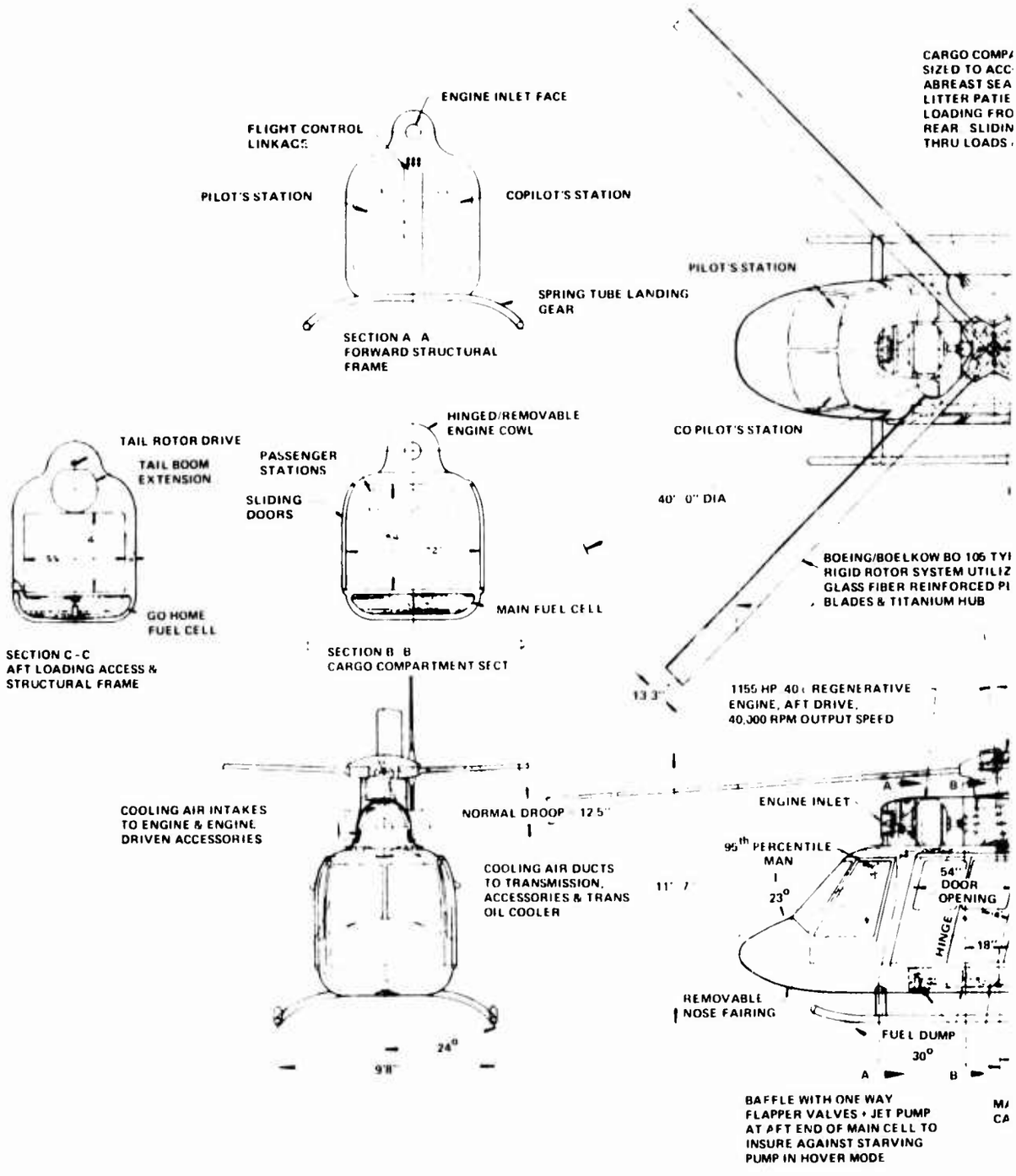
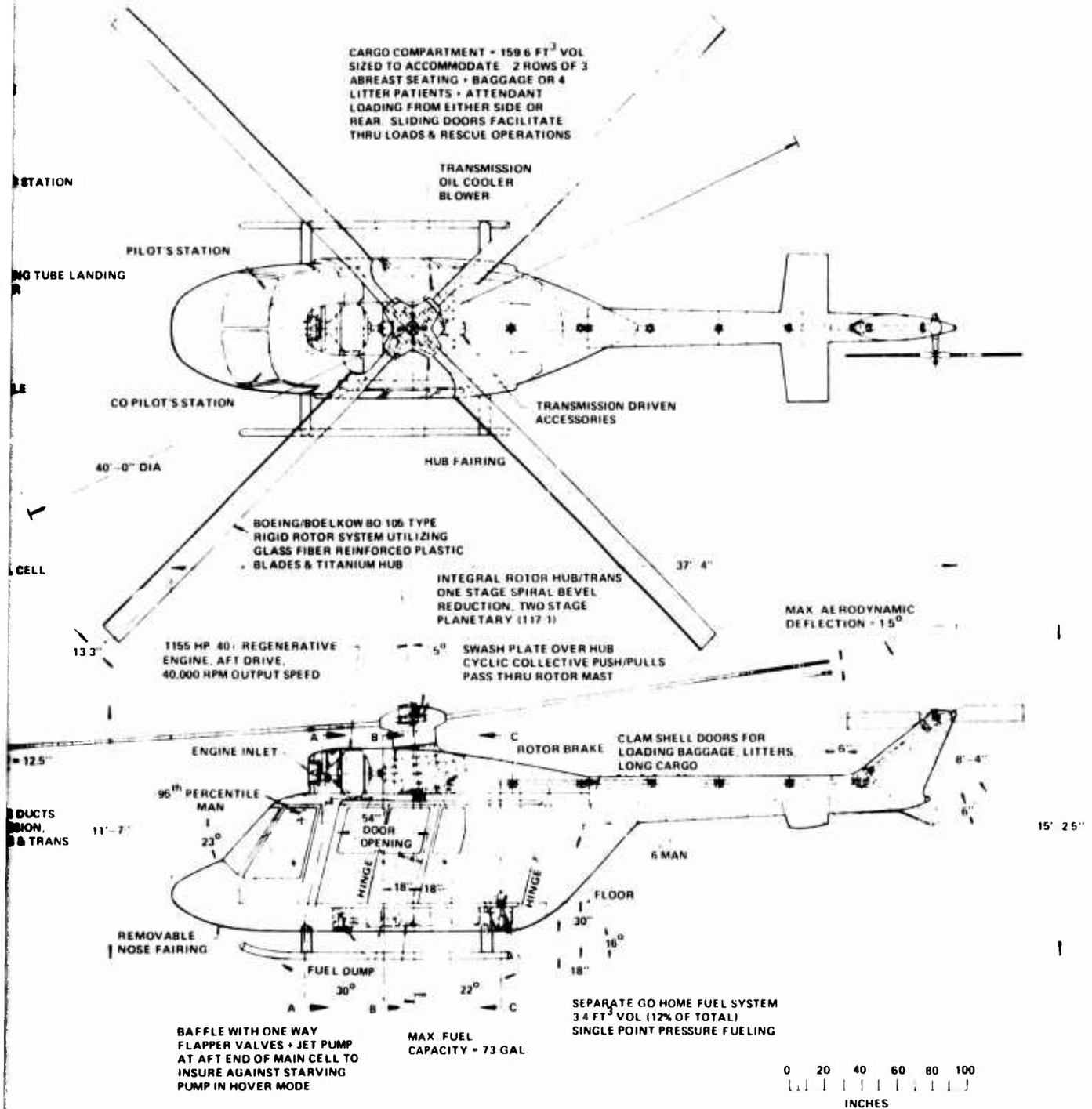


Figure 53. General Arrangement, Utility Helicopter With Single Advanced-Technology Regenerative Engine (0.40 Effectiveness).



ility Helicopter
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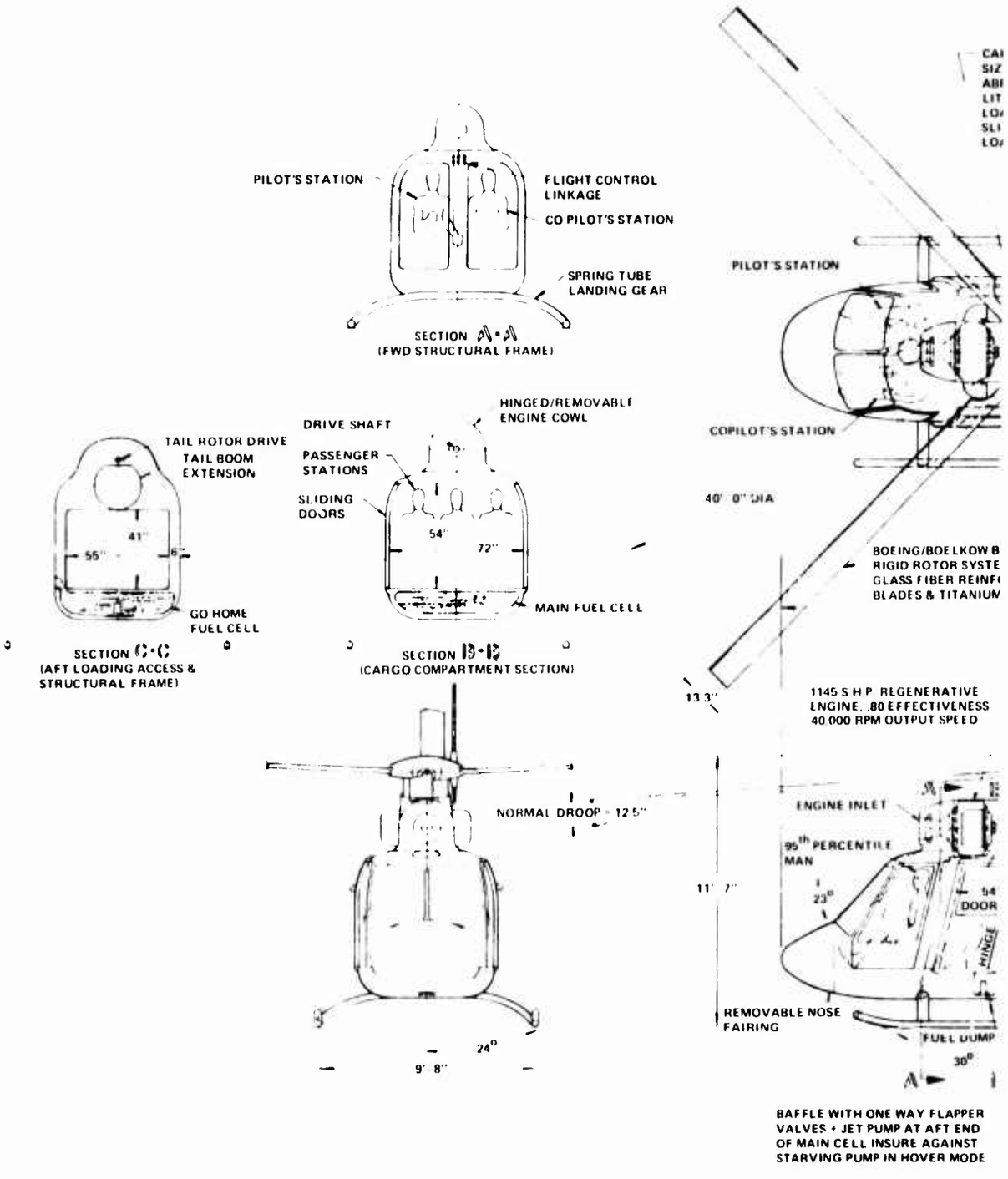
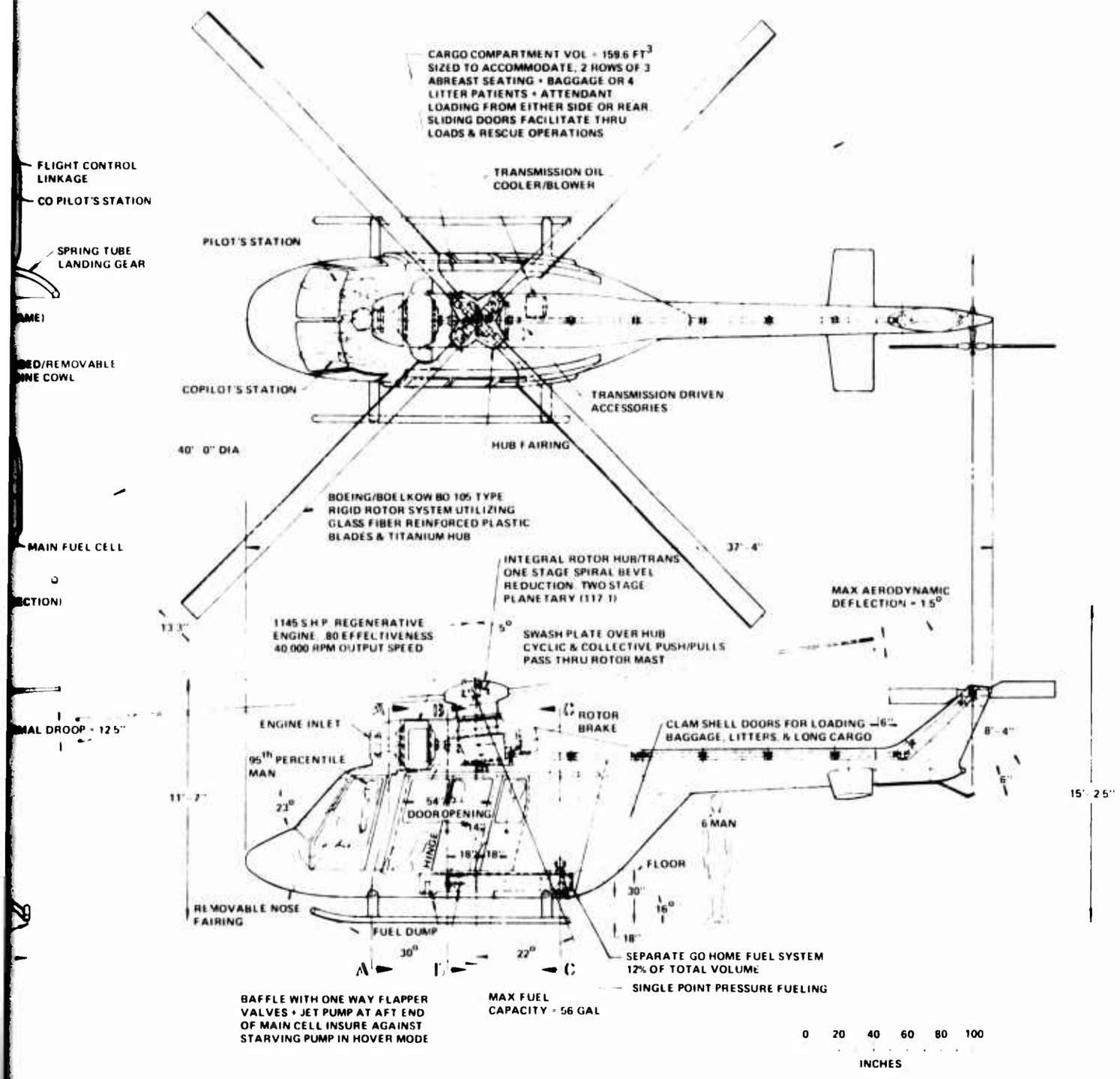


Figure 54. General Arrangement, Utility Helicopter With Single Advanced-Technology Regenerative Engine (0.80 Effectivenss).



Utility Helicopter
Technology Regenerative
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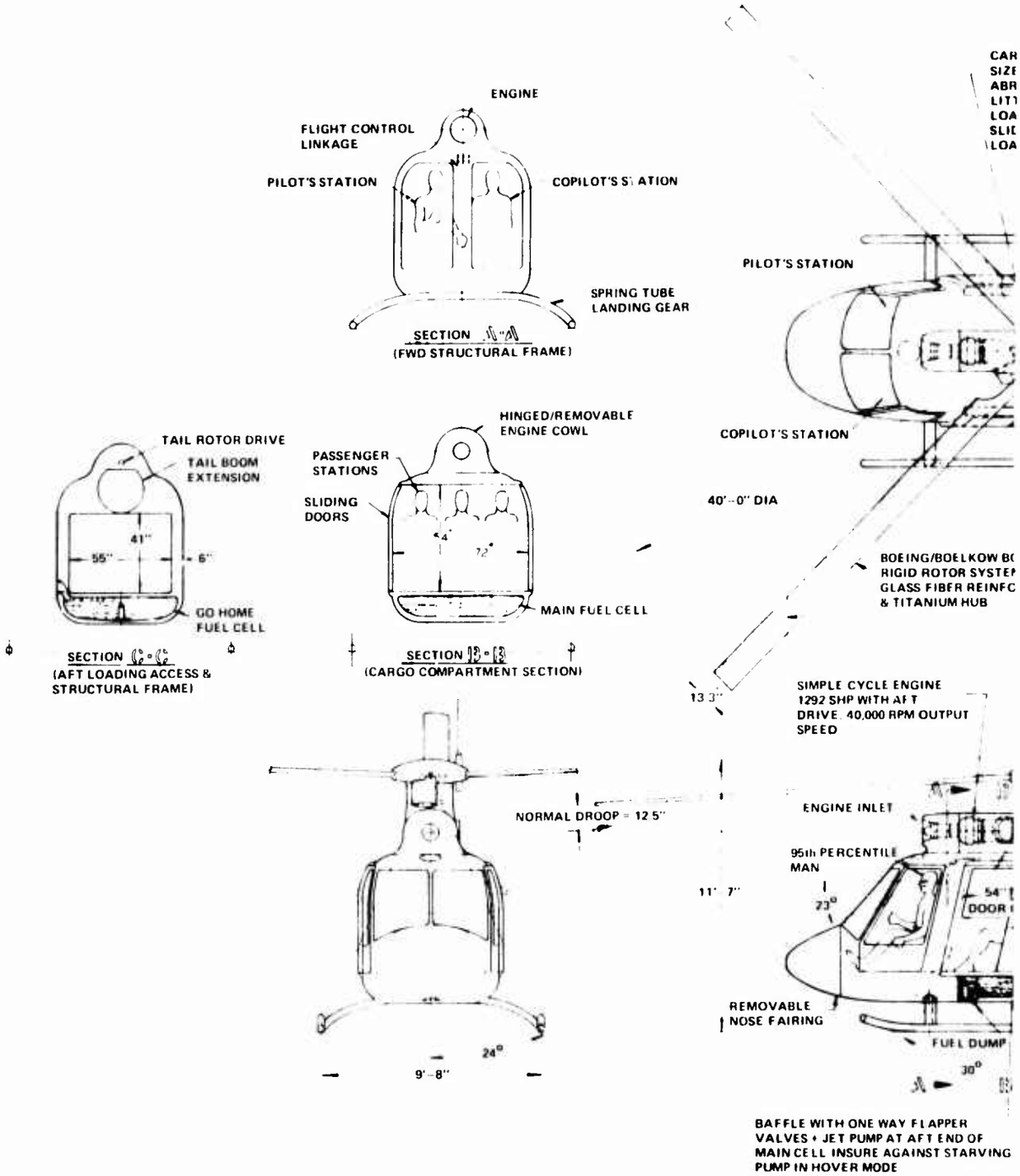
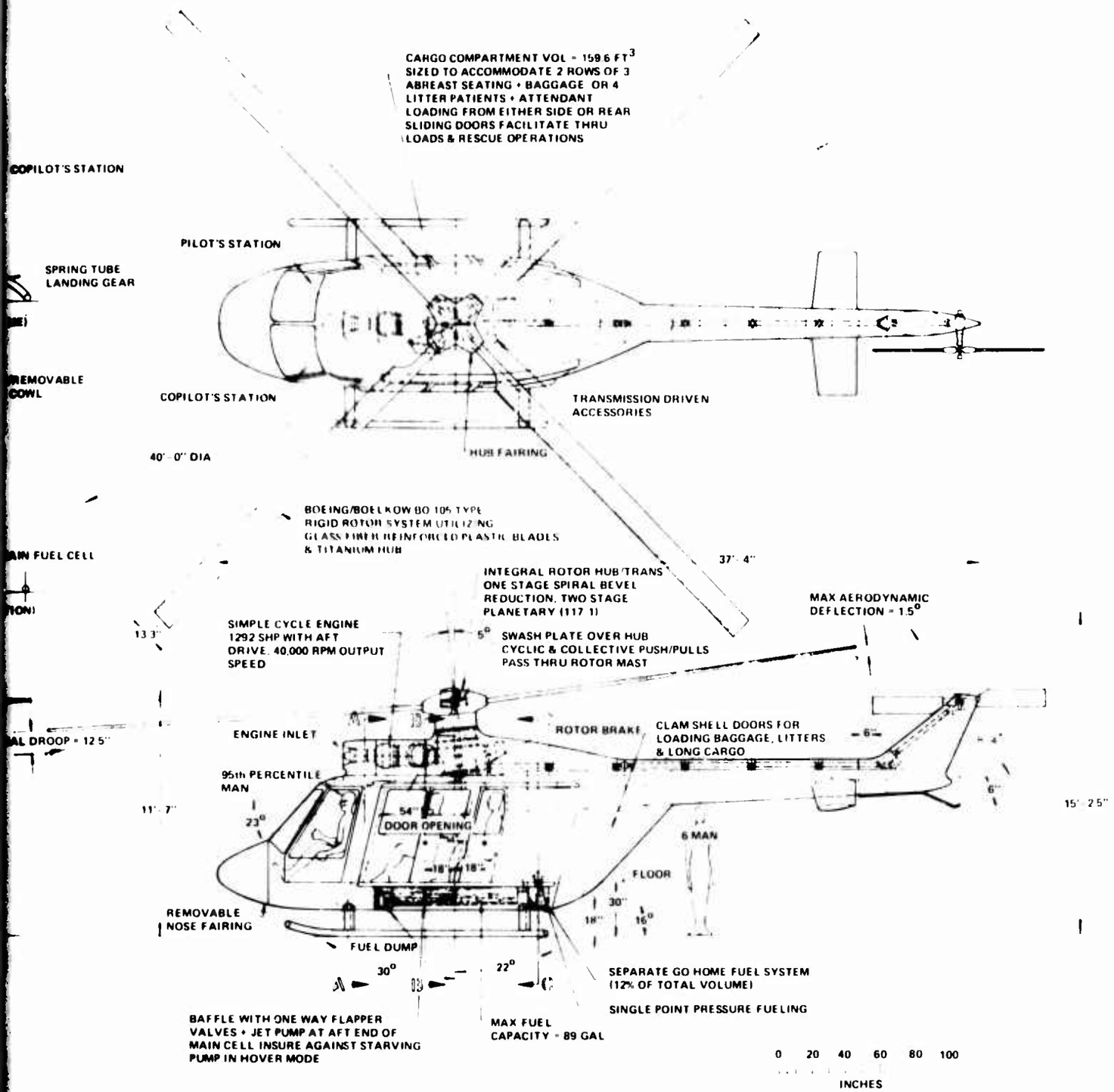


Figure 55. General Arrangement, Utility Helicopter With Single Available-Technology Simple-Cycle Engine.



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logy Simple-Cycle Engine.

for the horizontal tail-rotor drive. The pilot and copilot stations are cantilevered from the front of the box.

The aircraft configurations differ only in the cowl sizes (which were dictated by the dimensions of the various engines) and in the fuel tank capacity (which reflected the differences in engine SFC's).

AIRCRAFT COMPARATIVE ANALYSES

The five single-engine conceptual aircraft designs, powered by regenerative and nonregenerative engines, were based on existing and future Army mission requirements for a utility transport helicopter. Comparative analyses included weight and performance parameters, reliability aspects, maintenance requirements, life-cycle cost and cost-effectiveness for various missions, and overall mission effectiveness and system cost.

COMPARATIVE WEIGHT PARAMETERS

Design gross weight and empty weight trends are shown in Figure 56 for the four aircraft with advanced-technology engines (regenerative and simple-cycle). These curves show that the engine with the 0.65-effectiveness recuperator results in the minimum weight for the aircraft, and the weights were lower than the aircraft powered by a simple-cycle engine (effectiveness = 0). The differences in the gross weight and empty weight, as a function of effectiveness, are small, however, reflecting the marginal improvements in SFC which could be achieved with a regenerative engine over the optimum advanced-technology simple-cycle engine. The weight parameters for the aircraft powered by an available-technology engine, plotted as individual points in Figure 56, were substantially higher than values on these curves.

In Figure 57, the engine dry weight and the fuel weight for the design utility mission are shown as a function of recuperator effectiveness for these aircraft. As expected, the fuel weight decreased and the engine weight increased monotonically with increasing effectiveness for the advanced-technology engines. However, the increasing slope of the engine weight curve resulting from the rapid increase in recuperator size (required to achieve very high values of effectiveness) more than offset decreased fuel requirements at these high values of effectiveness. Consequently, the engine with the 0.65-effectiveness recuperator produced the minimum aircraft weights (Figure 56).

COMPARATIVE PERFORMANCE

Installed engine powers at the sea level, 59°F, Military Rated Power are shown in Figure 58. The transmission limit, which corresponds to the Military Rated Power at 4000 feet, 95°F, is

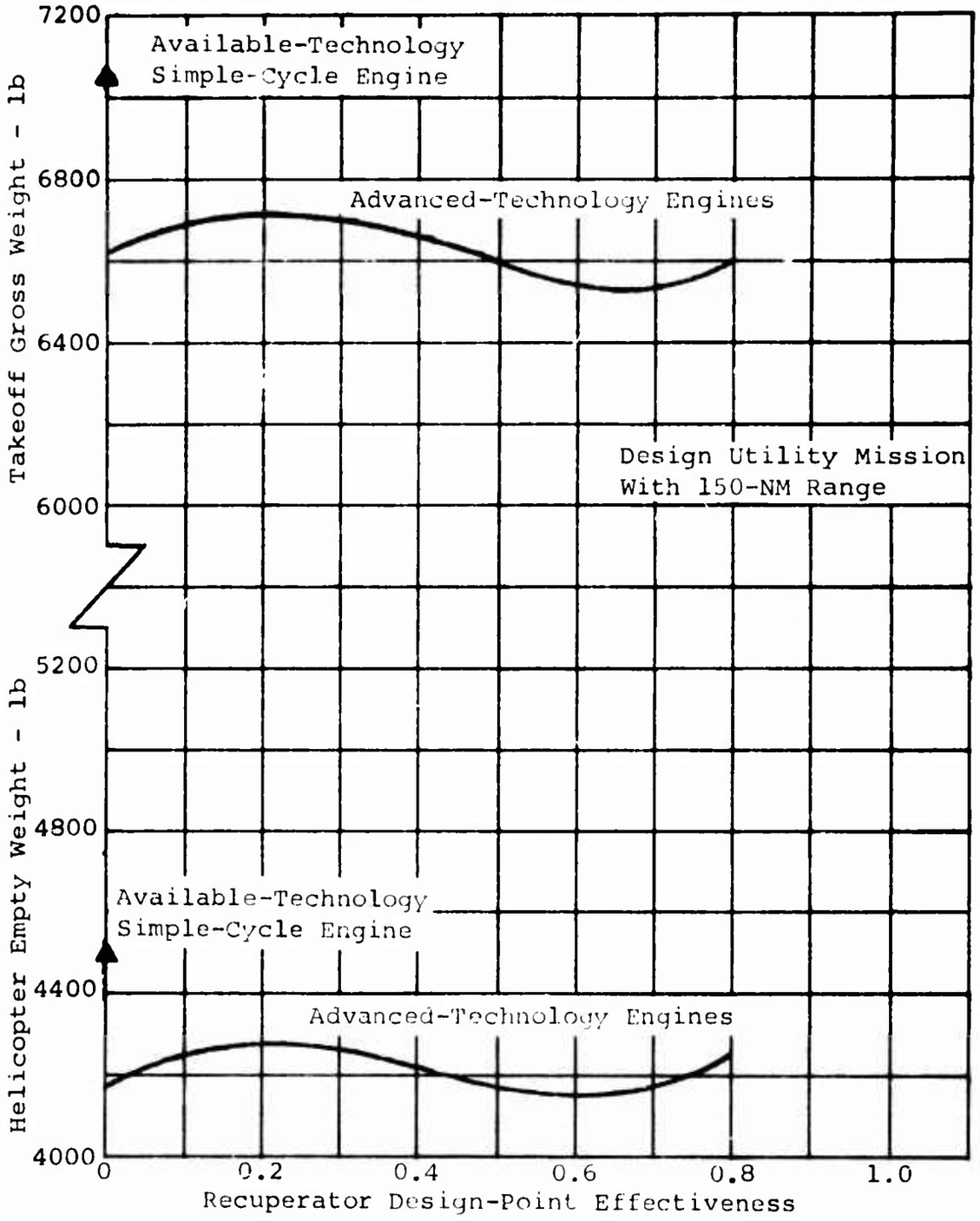


Figure 56. Takeoff Gross Weight and Empty Weight for Helicopters With Regenerative and Nonregenerative Engines.

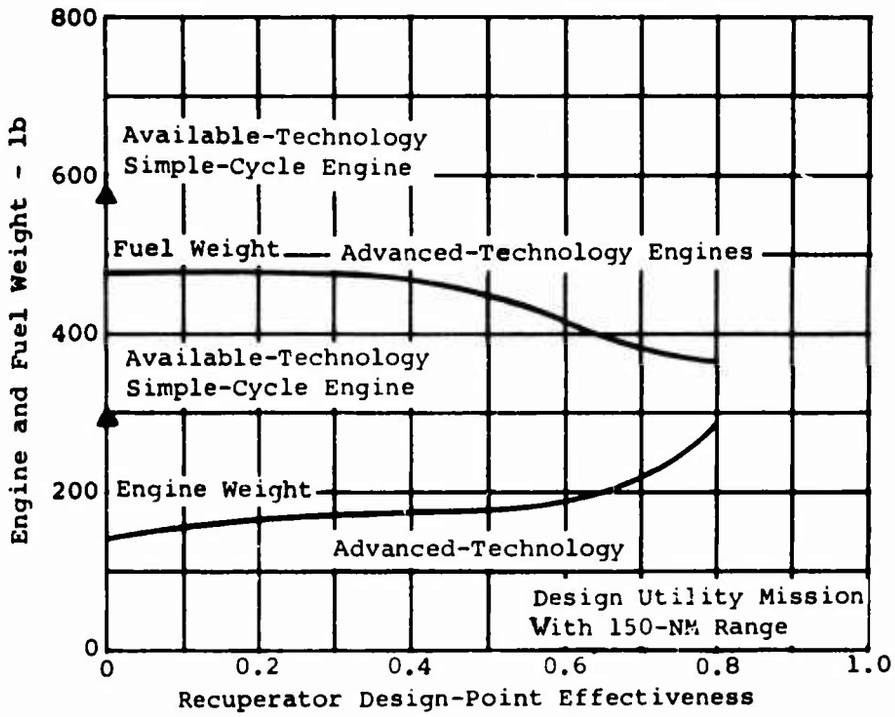


Figure 57. Engine and Fuel Weight as a Function of Effectiveness.

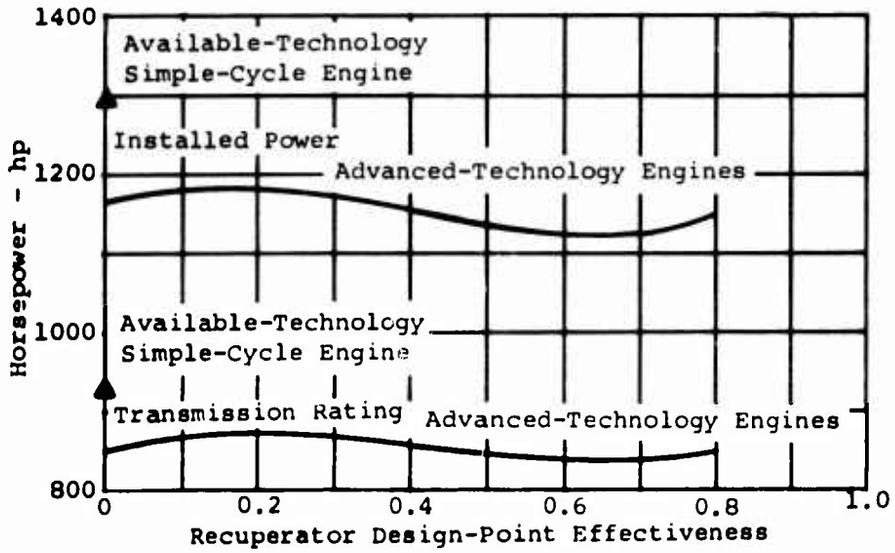


Figure 58. Installed Power and Transmission Rating as a Function of Effectiveness.

also given in Figure 58. The trends of these curves and plotted points reflect the aircraft gross weight trends previously plotted.

In Figure 59, the power required and power available are shown for aircraft cruise performance. The installed power available curves correspond to the Normal Rated Power of the engine at 4000 feet, 95°F. The cruise velocity for the utility mission was chosen at the NRP capability of the engine. Because the curves are virtually the same for all the aircraft with regenerative engines, only one is displayed in the figure.

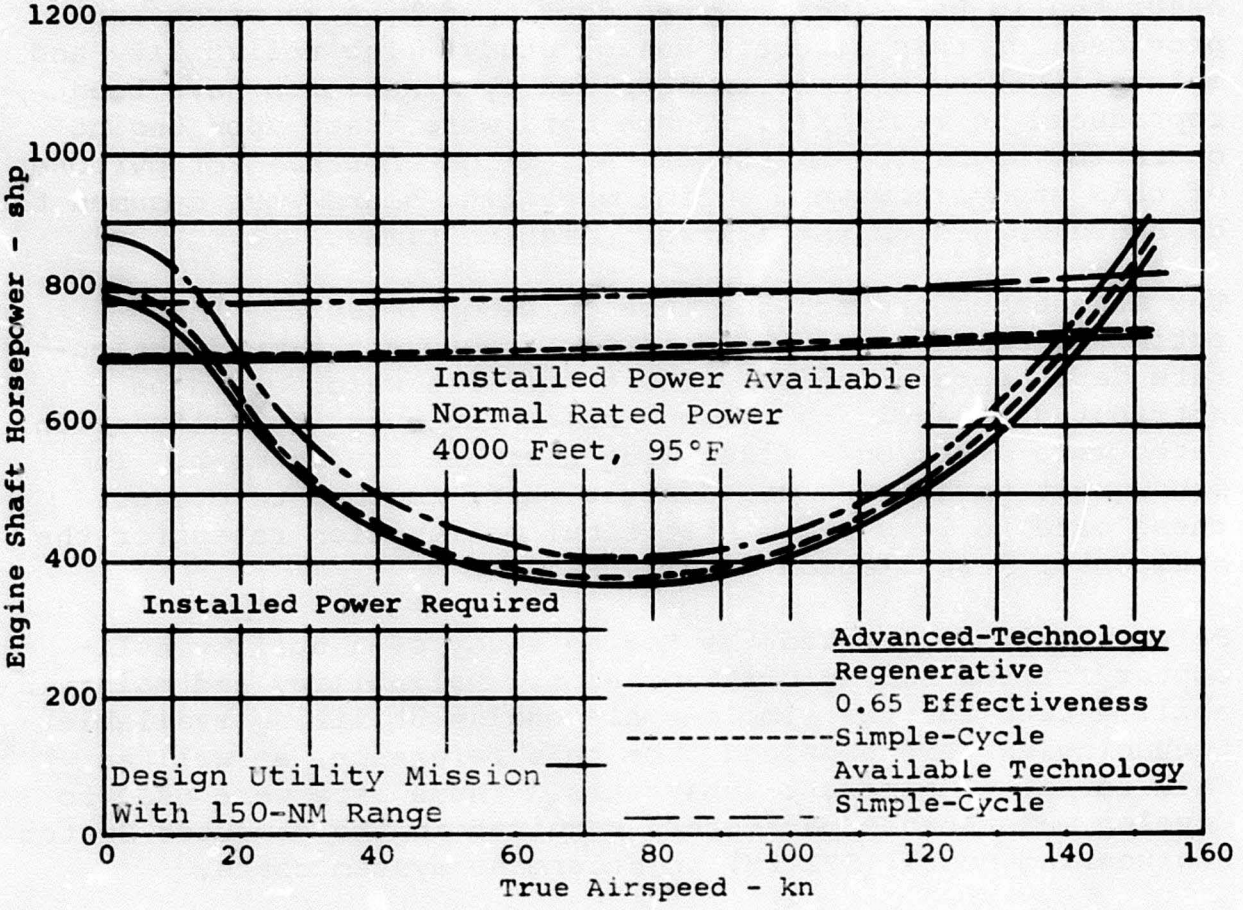


Figure 59. Power Required and Power Available Curves for Aircraft With Regenerative and Nonregenerative Engines.

Payload-range curves for all the aircraft are pictured in Figure 60 at the design altitude and ambient temperature. Although the helicopters with regenerative engines have lower fuel requirements for the design mission of 150 NM than those with simple-cycle engines, operation with simple-cycle engines over shorter distances would result in greater payloads, by off-loading fuel, than their regenerative engine-powered counterparts. This factor had a considerable impact on the cost-effectiveness studies, to be discussed later.

SUBSYSTEM RELIABILITY AND MAINTAINABILITY PARAMETERS

Reliability and maintainability data, consistent with the advanced-technology regenerative and nonregenerative engines described in Reference 1, were developed by AiResearch and provided for this aircraft design study. The reliability and maintainability parameters supplied by AiResearch have been reproduced in Table XII. These data were based upon engine operating hours (OHRS) rather than flight hours. For purposes of this study, however, engine operating hours were assumed to be synonymous with the aircraft mission times.

The AiResearch predictions were quite optimistic. This optimism could be justified in part because the malfunction-rate data represented only those failures which could be attributed directly to the engine and because the malfunction rates were based on a fixed-wing aircraft environment. In subsequent paragraphs the derivation of factors to convert these data to less optimistic total malfunction rates for the helicopter installation is discussed.

Reference 12 documented the Boeing study of a utility helicopter for the Army's UTTAS mission. Reliability and maintainability data for the simple-cycle engine utilizing available technologies were obtained from this reference, as well as data for other aircraft subsystems. These data were used to develop maintenance parameters required in the computer System Engineering Model (SYSTEM) to determine system costs.

Reliability Parameters

Maintenance-affecting and mission-affecting malfunction rates, total engine removal rates, and probabilities of mission completion were developed.

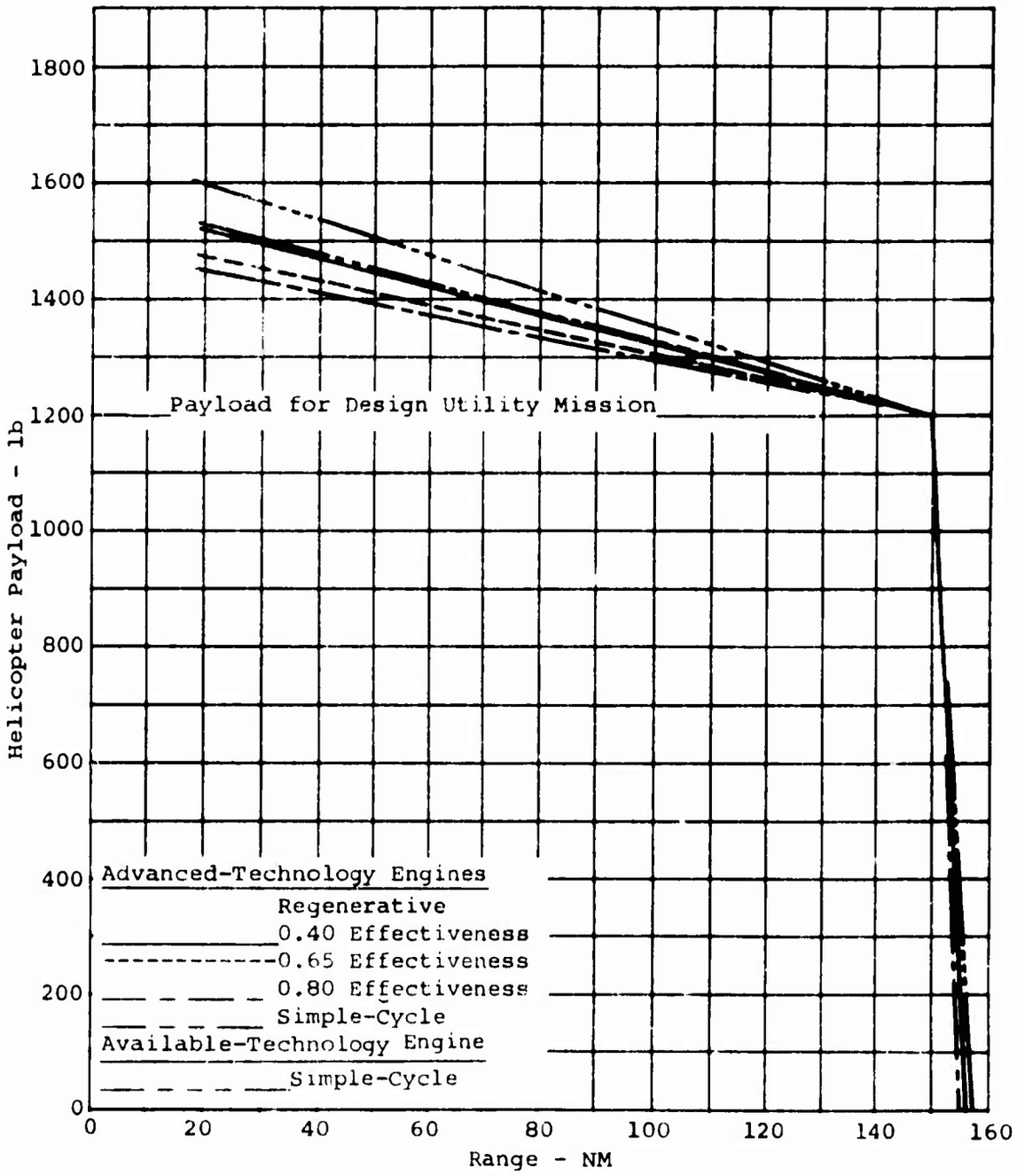


Figure 60. Payload-Range Curves for Helicopters With Regenerative and Nonregenerative Engines.

Maintenance-affecting malfunctions are defined as the sum of primary malfunctions and nonprimary malfunctions. A primary malfunction is defined as any malfunction occurring during the useful life of the component that is not caused by faulty maintenance, improper handling, improper operator technique, foreign object damage, or failure of related components. A nonprimary malfunction is defined as any malfunction that is caused by faulty maintenance, improper handling, improper operator technique, foreign object damage, or failure of related components.

Maintenance-affecting malfunction data for the available-technology engine were extracted from Reference 12. The data supplied by AiResearch, listed in Table XII, were used as a baseline for the advanced-technology engines. Since the latter data were for a fixed-wing aircraft environment, it was necessary to modify the malfunction rates to account for the more severe environment of the helicopter installation. A multiplication factor of 2.0 was used to represent the anticipated increase. The adjusted data, corresponding to primary malfunction rates for advanced-technology engines in the helicopter installation, are included in Table XIII.

A further adjustment was required to calculate total malfunction rates. Data for the T55-L-7C engine were used to determine the relationship between total malfunction rates (including primary and nonprimary malfunctions) and primary malfunction rates. The T55-L-7C engine is used in Boeing's CH-47 helicopter and represents a valid, well-understood data base. Maintenance-affecting total malfunction rates were developed using this historical base, which resulted in the following multiplication factors for occurrence of the replacement tasks:

1. For the first 10 items in Table XIII (fuel control replacement to power turbine governor replacement), a multiplication factor of 1.3 was used.

2. For the 5 seal replacements, a factor of 1.0 was used.

TABLE XII. PRELIMINARY MAINTAINABILITY DATA PROVIDED BY AiRESEARCH FOR REGENERATIVE AND NONREGENERATIVE ENGINES

Component	Task	Level	Crew Size	Regenerative Engines			Nonregenerative Engines		
				MMH (hr)	Occurrence 1000 hr	MMH 1000 hr	MMH (hr)	Occurrence 1000 hr	MMH 1000 hr
Unscheduled Corrective Maintenance									
Fuel Control	Replace	On-acft	1	1.000	1.000	1.000	1.000	1.000	1.000
Igniter Plug	Replace	On-acft	1	0.250	0.050	0.013	0.217	0.050	0.011
Fuel Nozzles	Replace	On-acft	1	0.600	0.300	0.180	0.600	0.300	0.180
Fuel Pump	Replace	On-acft	1	1.250	0.050	0.063	1.250	0.050	0.063
Pressurization/Dump Valve	Replace	On-acft	1	0.750	0.050	0.038	0.750	0.050	0.038
Oil Vent Valve	Replace	On-acft	1	0.500	0.025	0.013	0.500	0.025	0.013
Ignition Unit	Replace	On-acft	1	0.333	0.050	0.017	0.333	0.050	0.017
Torque Sensor	Replace	On-acft	1	0.417	0.200	0.084	0.417	0.200	0.084
Oil Transfer Tube Seal	Replace	On-acft	1	0.600	0.025	0.015	0.600	0.025	0.015
Power Turbine Governor	Replace	On-acft	1	0.750	0.050	0.038	0.600	0.050	0.030
Governor Mounting Pad Shaft Seal	Replace	On-acft	1	1.250	0.100	0.125	1.000	0.100	0.100
Power Output Shaft Seal	Replace	On-acft	1	0.583	0.100	0.058	0.583	0.100	0.058
Starter/Generator Pad Seal	Replace	On-acft	1	0.500	0.100	0.050	0.500	0.100	0.050
Rear Turbine Seal	Replace	On-acft	1	1.750	0.125	0.219	1.750	0.125	0.219
Combustor	Replace	On-acft	2(a)	7.000	0.050	0.350	3.000	0.050	0.150
T_4 Temp Sensor	Replace	On-acft	2(a)	7.500	0.050	0.375	3.500	0.050	0.175
Front Compressor Seal	Replace	Field Shop	1	2.500	0.125	0.313	2.500	0.125	0.313
Recuperator Module	Replace	Field Shop	1	2.500	0.050	0.125	-	-	-
Gearbox	Replace	Depot	1	2.500	0.100	0.250	2.500	0.100	0.250
Gas Generator Module	Replace	Depot	1	5.000	0.200	1.000	5.000	0.200	1.000
Power Turbine Module	Replace	Depot	1	5.500	0.200	1.100	5.500	0.200	1.100
Gas Generator Module	Overhaul	Depot	1	16.000	0.200	3.200	16.000	0.200	3.200
Power Turbine Module	Overhaul	Depot	1	12.000	0.200	2.400	12.000	0.200	2.400
Recuperator Module	Repair	Depot	1	8.000	0.050	0.400	-	-	-
Fuel Control	Overhaul	Depot	1	12.000	1.00	12.00	12.000	1.00	12.00
Fuel Nozzles	Overhaul	Depot	1	4.000	0.300	1.200	4.000	0.300	1.200
Gearbox	Overhaul	Depot	1	8.000	0.100	0.800	8.000	0.100	0.800
Fuel Pump	Overhaul	Depot	1	4.000	0.050	0.200	4.000	0.050	0.200
Pressurization/Dump Valve	Overhaul	Depot	1	6.000	0.050	0.300	6.000	0.050	0.300
Oil Vent Valve	Overhaul	Depot	1	4.000	0.025	0.100	4.000	0.025	0.100
Torque Sensor	Overhaul	Depot	1	3.000	0.200	0.600	3.000	0.200	0.600
Power Turbine Governor	Overhaul	Depot	1	3.500	0.050	0.175	3.500	0.050	0.175
Combustor	Repair	Depot	1	1.500	0.050	0.075	1.500	0.050	0.075

Starter/Generator Pad Seal	**Replace**	**On-acft**	**1**	**0.500**	**0.100**	**0.050**	**0.500**	**0.100**	**0.050**
Rear Turbine Seal	Replace	On-acft	1	1.750	0.125	0.219	1.750	0.125	0.219
Combustor	Replace	On-acft	2(a)	7.000	0.050	0.350	3.000	0.050	0.150
T_4 Temp Sensor	Replace	On-acft	2(a)	7.500	0.050	0.375	3.500	0.050	0.175
Front Compressor Seal	Replace	Field Shop	1	2.500	0.125	0.313	2.500	0.125	0.313
Recuperator Module	Replace	Field Shop	1	2.500	0.050	0.125	-	-	-
Gearbox	Replace	Depot	1	2.500	0.100	0.250	2.500	0.100	0.250
Gas Generator Module	Replace	Depot	1	5.000	0.200	1.000	5.000	0.200	1.000
Power Turbine Module	Replace	Depot	1	5.500	0.200	1.100	5.500	0.200	1.100
Gas Generator Module	Overhaul	Depot	1	16.000	0.200	3.200	16.000	0.200	3.200
Power Turbine Module	Overhaul	Depot	1	12.000	0.200	2.400	12.000	0.200	2.400
Recuperator Module	Repair	Depot	1	8.000	0.050	0.400	-	-	-
Fuel Control	Overhaul	Depot	1	12.000	1.00	12.00	12.000	1.00	12.00
Fuel Nozzles	Overhaul	Depot	1	4.000	0.300	1.200	4.000	0.300	1.200
Gearbox	Overhaul	Depot	1	8.000	0.100	0.800	8.000	0.100	0.800
Fuel Pump	Overhaul	Depot	1	4.000	0.050	0.200	4.000	0.050	0.200
Pressurization/Dump Valve	Overhaul	Depot	1	6.000	0.050	0.300	6.000	0.050	0.300
Oil Vent Valve	Overhaul	Depot	1	4.000	0.025	0.100	4.000	0.025	0.100
Torque Sensor	Overhaul	Depot	1	3.000	0.200	0.600	3.000	0.200	0.600
Power Turbine Governor	Overhaul	Depot	1	3.500	0.050	0.175	3.500	0.050	0.175
Combustor	Repair	Depot	1	1.500	0.050	0.075	1.500	0.050	0.075
Scheduled Preventive Maintenance									
Fuel System	**Change Fuel Filter**	**On-acft**	**1**	**0.083**	**5.000**	**0.415**	**0.083**	**5.000**	**0.415**
Oil System	**Change Oil Filter**	**On-acft**	**1**	**0.083**	**5.000**	**0.415**	**0.083**	**5.000**	**0.415**
Gas Turbine Engine	**Inspect**	**On-acft**	**1**	**0.133**	**20.000**	**2.666**	**0.083**	**20.000**	**1.666**
Gas Turbine Engine	**Overhaul**	**Depot**	**1**	**55.000**	**1.000(b) (initial)**	**55.000**	**50.000**	**1.000(b) (initial)**	**50.000**
Gas Turbine Engine	**Replenish Oil**	**On-acft**	**1**	**0.083**	**20.000**	**1.660**	**0.083**	**20.000**	**1.660**
Gas Turbine	**Change Oil**	**On-acft**	**1**	**0.167**	**5.000**	**0.835**	**0.167**	**5.000**	**0.835**

(a) Crew Size is (1) Man for nonregenerative engine.

(b) As shown above, it is anticipated that the initial TBO will be no less than 1000 OHRS with normal growth to 2000 OHRS or more.

TABLE XIII. ADJUSTED MAINTAINABILITY DATA FOR REGENERATIVE AND NONREGENERATIVE ENGINES

Component	Task	Level	Crew	Regenerative Engines			Nonregenerative Engines		
				MMH (hr)	Occurrence 1000 hr	MMH 1000 hr	MMH (hr)	Occurrence 1000 hr	MMH 1000 hr
Unscheduled Corrective Maintenance									
Fuel Control	Replace	On-acft	1	1.450	2.000	2.900	1.450	2.000	2.900
Igniter Plug	Replace	On-acft	1	0.450	0.100	0.045	0.350	0.100	0.035
Fuel Nozzles	Replace	On-acft	1	0.717	0.600	0.430	0.700	0.600	0.420
Fuel Pump	Replace	On-acft	1	1.800	0.100	0.180	1.920	0.100	0.192
Pressurization/Dump Valve	Replace	On-acft	1	1.050	0.100	0.105	1.100	0.100	0.110
Oil Vent Valve	Replace	On-acft	1	0.610	0.050	0.031	0.610	0.050	0.031
Ignition Unit	Replace	On-acft	1	0.390	0.100	0.039	0.390	0.100	0.039
Torque Sensor	Replace	On-acft	1	0.475	0.400	0.190	0.475	0.400	0.190
Oil Transfer Tube Seal	Replace	On-acft	1	0.750	0.050	0.038	0.700	0.050	0.035
Power Turbine Governor	Replace (a)	On-acft	1	0.750	0.100	0.075	0.750	0.100	0.075
Governor Mounting Pad Shaft Seal	Replace (a)	On-acft	1	1.150	0.200	0.230	1.150	0.200	0.230
Power Output Shaft Seal	Replace (b)	On-acft	2	2.143	0.100	0.214	2.088	0.100	0.209
Starter/Generator Pad Seal	Replace	On-acft	1	0.600	0.200	0.120	0.600	0.200	0.120
Rear Turbine Seal	Replace (b)	On-acft	2	3.310	0.125	0.414	3.255	0.125	0.407
Combustor	Replace (c)	Field Shop	2 (d)	7.250(e)	0.050	0.363	3.200	0.050	0.160
T_4 Temp Sensor	Replace (c)	Field Shop	2 (d)	7.650(e)	0.050	0.383	3.700	0.050	0.185
Front Compressor Seal	Replace (c)	Field Shop	1	2.550	0.125	0.319	2.550	0.125	0.319
Recuperator Module	Replace (c)	Field Shop	2	2.600(e)	0.050	0.130	-	-	-
Gearbox	Replace	Depot	1	5.000	0.100	0.500	5.000	0.100	0.500
Gas Generator Module	Replace	Depot	1	10.000	0.200	2.000	10.000	0.200	2.000
Power Turbine Module	Replace	Depot	1	13.550(f)	0.200	2.710	11.000	0.200	2.200
Gas Generator Module	Overhaul	Depot	1	32.000	0.200	6.400	32.000	0.200	6.400
Power Turbine Module	Overhaul	Depot	1	24.000	0.200	4.800	24.000	0.200	4.800
Recuperator Module	Repair	Depot	1	16.000	0.050	0.800	-	-	-
Fuel Control	Overhaul	Depot	1	24.000	2.000	48.000	24.000	2.000	48.000
Fuel Nozzles	Overhaul	Gen. Supp.	1	8.000	0.600	4.800	8.000	0.600	4.800
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Rear Turbine Seal	Replace (b)	On-acft	2	3.310	0.125	0.414	3.200	0.125	0.400
Combustor	Replace (c)	Field Shop	2 (d)	7.250(e)	0.050	0.363	3.200	0.050	0.160
T_4 Temp Sensor	Replace (c)	Field Shop	2 (d)	7.650(e)	0.050	0.383	3.700	0.050	0.185
Front Compressor Seal	Replace (c)	Field Shop	1	2.550	0.125	0.319	2.550	0.125	0.319
Recuperator Module	Replace (c)	Field Shop	2	2.600(e)	0.050	0.130	-	-	-
Gearbox	Replace	Depot	1	5.000	0.100	0.500	5.000	0.100	0.500
Gas Generator Module	Replace	Depot	1	10.000	0.200	2.000	10.000	0.200	2.000
Power Turbine Module	Replace	Depot	1	13.550(f)	0.200	2.710	11.000	0.200	2.200
Gas Generator Module	Overhaul	Depot	1	32.000	0.200	6.400	32.000	0.200	6.400
Power Turbine Module	Overhaul	Depot	1	24.000	0.200	4.800	24.000	0.200	4.800
Recuperator Module	Repair	Depot	1	16.000	0.050	0.800	-	-	-
Fuel Control	Overhaul	Depot	1	24.000	2.000	48.000	24.000	2.000	48.000
Fuel Nozzles	Overhaul	Gen. Supp.	1	8.000	0.600	4.800	8.000	0.600	4.800
Gearbox	Overhaul	Depot	1	16.000	0.100	1.600	16.000	0.100	1.600
Fuel Pump	Overhaul	Gen. Supp.	1	8.000	0.100	0.800	8.000	0.100	0.800
Pressurization/Dump Valve	Overhaul	Depot	1	12.000	0.100	1.200	12.000	0.100	1.200
Oil Vent Valve	Overhaul	Gen. Supp.	1	8.000	0.050	0.400	8.000	0.050	0.400
Torque Sensor	Overhaul	Depot	1	6.000	0.400	2.400	6.000	0.400	2.400
Power Turbine Governor	Overhaul	Depot	1	7.000	0.100	0.700	7.000	0.100	0.700
Combustor	Repair	Gen. Supp.	1	3.000	0.050	0.150	3.000	0.050	0.150
Scheduled Preventive Maintenance									
Fuel System	Change Fuel Filter	On-acft	1	0.133	5.000	0.665	0.133	5.000	0.665
Oil System	Change oil Filter	On-acft	1	0.138	5.000	0.690	0.138	5.000	0.690
Gas Turbine Engine	Inspect	On-acft	1	0.183	20.000	3.660	0.133	20.000	2.660
Gas Turbine Engine	Overhaul	Depot	2	204.000	1.000(g) (initial)	204.000	268.000	1.000(g) (initial)	268.000
Servicing (Scheduled)									
Gas Turbine Engine	Replenish Oil	On-acft	1	0.133	20.000	2.660	0.133	20.000	2.660
Gas Turbine	Change oil	On-acft	1	0.300	5.000	1.500	0.300	5.000	1.500

(a) These items are transmission-mounted - no difference in task time between regenerative and nonregenerative.
(b) Engine removal is required for this task. MMH required are included in task.
(c) Engine removal is required for this task. MMH to remove and replace engine included as engine unscheduled removals.
(d) Crew size is 1 man for nonregenerative engine.
(e) Task times on these items vary slightly according to regenerator effectiveness. Shown are times for 0.45 effectiveness regenerator.
(f) This includes time to remove and replace an in-the-way component - the recuperator module.
(g) As shown above, it is anticipated that the Initial TBO will be no less than 1000 HRS with normal growth to 2000 HRS or more.

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3. For the replacement of the combustor, T_4 temperature sensor, and recuperator module, a factor of 2.3 was used.

4. For the replacement of the gearbox, gas generator module, and power turbine module, a factor of 2.6 was used.

The overhaul interval (TBO) was specified as 1000 hours. Engines not removed earlier (not involved in a premature component failure, which otherwise would require removal for unscheduled maintenance) still would be removed at the 1000-hour point for scheduled overhaul. Unscheduled engine removal rates were derived from the maintenance-affecting malfunction rates in Table XIII, with the additional correction applied for the relationships between unscheduled removals and maintenance-affecting malfunctions derived from T55-L-7C experience. Total engine removals are the sum of scheduled and unscheduled engine removals. The reciprocal of the total engine removal rate is the Mean Time Between Removal (MTBR).

> Mission-affecting malfunctions were defined as those maintenance-affecting malfunctions which cause a scheduled mission to be aborted, cancelled, or delayed (in excess of 15 minutes) after the aircraft has been declared operationally ready.

Predicted rates for the advanced-technology nonregenerative and regenerative engines were derived from the relationship of total removal rates to mission-affecting malfunctions, based on AiResearch engine experience.

Probability of mission completion was calculated for a mission duration of 1 hour. The resulting value represents the probability of starting and completing the mission of 1-hour duration with an aircraft declared operationally ready without a cancelled takeoff, a takeoff delayed in excess of 15 minutes, or an in-flight abort caused by a mission-affecting malfunction of the engine.

The resulting data are presented in Table XIV. The maintenance-affecting malfunction rates include primary and nonprimary malfunctions for the engine installed in the helicopter operating environment. It can be seen that significant improvements in malfunction and removal rates

TABLE XIV. RELIABILITY PARAMETERS FOR REGENERATIVE AND NONREGENERATIVE ENGINES INSTALLED IN A UTILITY TRANSPORT HELICOPTER

	Regenerative Engines			Nonregenerative Engines	
	Effectiveness			Advanced-Technology	Available-Technology
	0.40	0.65	0.80		
Maintenance-Affecting Malfunctions					
Malfunctions/Flight Hour-1/hour	.00780	.00780	.00780	.00767	.01040
Mean-Time-Between-Malfunction-Hr	128.	128.	128.	131.	96.
Mission-Affecting Malfunctions					
Malfunctions/Flight Hour-1/hr	.000176	.000176	.000176	.000164	.000200
Total Engine Removals					
Removals/Flight Hour-1/hr	.00200	.00200	.00200	.00195	.00218
Mean-Time-Between Removals-Hr	500.	500.	500.	513.	429.
Probability of Mission Completion (1-Hour Mission) - %	99.9824	99.9824	99.9824	99.9838	99.9800

are predicted in going from available-technology simple-cycle engines to advanced-technology engines. Incorporating a recuperator into an advanced-technology engine results in an insignificant degradation in malfunction and removal rates. This degradation is considered to be independent of the regenerator effectiveness selected.

Maintainability Parameters

The maintainability parameters (MMH) provided by AiResearch for the advanced-technology regenerative and nonregenerative engines were adjusted for the aircraft design study, and the adjusted values were listed in Table XIII. The changes from the preliminary data were developed according to the following considerations:

1. For the on-aircraft repairs, the preliminary maintenance man-hours were adjusted to include engine accessibility considerations. An estimate was made of the time required to reach the engine area on the aircraft with the necessary tools and parts, open the cowling, gain access to the particular repair area, perform the required operation (adjusting the time according to the size and shape of the work area), secure the cowling, and leave the aircraft. Because of the need for accessibility, engine removal was necessary to accomplish some repairs which are indicated as on-aircraft actions in the preliminary data, and this consideration is noted in Table XIII.

2. For maintenance operations at the organizational or direct support level, the time required to obtain and/or transfer parts, tools, and equipment from their storage area to the aircraft and back would vary greatly and would be a function of the distance to the shop or parts room. Other considerations include the weather (if not operating on a hard-surfaced ramp) and the amount or type of ground equipment available. Because of the potentially large variations in time, these factors were not included in the MMH at the operational and direct support levels.

3. Several tasks not included in the preliminary maintainability data were added, and these tasks are included in the final maintenance figures. A daily visual inspection was assumed, based on an average of three flight hours for each inspection. MMH for the 50-hour visual inspection in Table XIII (11 minutes for regenerative engines and 8 minutes for nonregenerative engines) were based on the AiResearch data in Table XII plus 3 minutes to open and close the cowling. For the advanced-technology engines, it was assumed that the daily visual inspection required slightly less time than this 50-hour inspection. Maintenance times and frequencies also were calculated for trouble-shooting, adjustments, retorquing, correcting leaks in lines and hoses, and tightening bolts. An unscheduled hot-end inspection was considered, and the task time was predicated on removing the engine, performing the inspection in the local field shop, and returning the engine to service or to the overhaul depot, as determined by the inspection.

4. Unscheduled engine removals were considered, including removals required to accomplish repairs and for depot-replaceable modules, as noted in Table XIII. Engine teardown and buildup were included for both scheduled and unscheduled removals. Times for the tasks in paragraph 3 above were based on those for Boeing production helicopters, scaled down proportionately and adjusted for airframe differences. For all the repairs accomplished in the field shop, it was assumed that the engine was returned to service on the next aircraft requiring an engine.

5. Component overhaul and repair times were adjusted to include cleaning, inspection, functional testing and time to obtain parts, as well as an allowance for indirect personnel. Some items in Table XIII were specified to be repaired or overhauled at the general support level, as are similar items in the USAAVSCOM Standard Maintenance Allocation Chart. Engine overhaul maintenance man-hours provided by AiResearch were thought to be very optimistic. Consequently, overhaul hours for the advanced-technology simple-cycle engines were revised using as a basis data for contemporary production engines (whose overhaul

times included indirect personnel man-hours). This was accomplished by reducing the overhaul hours for the production engines in direct proportion to the ratio of the rated shaft powers of the advanced-technology and contemporary simple-cycle engines. Suitable adjustments were made for the modular design and light weight of the advanced-technology engines. To this adjusted value for the nonregenerative engine, the man-hours for repair of the recuperator module were added to obtain the overhaul time for the regenerative engines.

Maintainability parameters for advanced-technology regenerative and nonregenerative engines and for available-technology simple cycle engines are summarized in Table XV. The data for the advanced-technology engines incorporate the adjustments employed in Table XIII plus the numerical factors to convert primary malfunction rates to total malfunction rates, and also incorporate those few items mentioned in paragraph 3 above which were not included in Table XIII. Maintainability parameters developed in Reference 12 were assumed to represent the available-technology simple-cycle engine. No adjustments were made in these data, and the maintenance numbers were taken directly from that document.

TABLE XV. SUMMARY MAINTAINABILITY PARAMETERS FOR REGENERATIVE AND NONREGENERATIVE ENGINES

Engine Type	Maintenance Occurrences per Flight Hour	Maintenance Man-Hours/Flight Hour: Organizational, Direct Support, and General Support Levels	Maintenance Man-Hours/Flight Hour: Depot Level	Mean Time to Repair (hr)	Flight Hours Between Inspections (hr)
Available-Technology Simple-Cycle Engine	.01040	.195	1.045	1.350	100
Advanced Technology Simple-Cycle Engine	.00767	.112	.874	.813	50
Regenerative Engine .40 Effectiveness	.00780	.133	.923	.808	50
Regenerative Engine .65 Effectiveness	.00780	.133	.923	.808	50
Regenerative Engine .80 Effectiveness	.00780	.134	.923	.808	50

Mean-time-to-repair (MTTR) numbers for the advanced-technology engines were derived from the data in Table XIII. MTTR values for the available-technology engine were derived from Reference 12. These MTTR values represented elapsed time only for those items to be repaired on the aircraft. The MTTR for the nonregenerative advanced-technology engine was higher than for the regenerative engines. The total elapsed repair time and maintenance occurrences for the simple-cycle engine were less, primarily because it had no recuperator module to malfunction or fail. However, they did not decrease in the same proportion. Maintenance occurrences per flight hour were developed directly from the data in Table XIII.

The interval between inspections for the advanced-technology engines was established as 50 hours by AiResearch. For the available-technology engines, which included a failure detection system, no inspection was included between the daily and 100-hour inspections (Reference 12). Because there was no intermediate inspection, the daily and 100-hour inspections were more comprehensive, and every fourth 100-hour inspection was assumed to be even more detailed. For each 100-hour inspection, 2.52 man-hours were expended, and 17.65 man-hours were expended on every fourth 100-hour inspection. These data were included in the maintainability parameters for the powerplant.

Both the regenerative and nonregenerative advanced-technology engines had an initial TBO of 1000 hours - anticipated to increase to 2000 hours with normal growth. These engines had no scheduled hot-end inspection, and each scheduled 50-hour inspection was a visual inspection only, with the required man-hours as displayed in Table XIII.

Maintenance occurrences and maintenance man-hours per flight hour, mean time to repair, and flight hours between inspections have been developed for regenerative and nonregenerative engines in Table XV. Percentage of deferred maintenance actions, number of men assigned, downtime per flight hour for spares, and desired level of availability were not considered in this study, inasmuch as these items are governed by maintenance concepts and policies.

Maintainability parameters for aircraft subsystems other than the engine were obtained from Reference 12, and these data are summarized in Table XVI.

TABLE XVI. SUMMARY MAINTAINABILITY PARAMETERS FOR UTILITY HELICOPTER SUBSYSTEMS OTHER THAN THE ENGINE

	Maintenance Man-Hours / Flight Hour
Operational, Direct Support Levels	
Corrective Maintenance	1.632
Scheduled Maintenance	1.239
General Support Levels	0.199
Depot Level	
Structure	0.473
Minor Repairables*	0.226
Major Repairables**	2.350

* Minor repairables (on-condition components) include actuators, pumps and motors.

** Major repairables, less engines, include major dynamic components.

AIRCRAFT SYSTEM COST COMPARISONS

Life-cycle cost, cost-effectiveness for the design mission and secondary missions, and overall mission cost-effectiveness were calculated for the five design-point aircraft.

Parametric Life-Cycle Cost Prediction

The computerized engineering cost model is predicated upon the assumptions in the following paragraphs.

Research, Development, Test, and Engineering (RDTE)

Nonrecurring engineering design and nonrecurring tooling costs were based on the weight of 19 subsystems. Recurring prototype costs (preproduction test aircraft) were determined as follows:

1. Airframe cost was based on the weight of 19 subsystems.

2. Engine costs were based on power.

3. Avionics costs were constant input/output. Prototype spares were calculated as a percentage of recurring prototype costs (20 percent for airframe, 50 percent for engines, and 40 percent for avionics).

Component testing was the accumulated cost of the components and engineering labor consumed during the bench testing program for each subsystem. Fatigue test article costs were estimated as a fraction (80 percent) of the cost of the structure and alighting gear portion of the first aircraft produced, as were the costs of the static test articles.

Flight testing costs were based on the number of flight test hours in the flight test program and all non-recurring (preparation and instrumentation - $5,600,000) flight-test costs.

Class I mock-ups were estimated as a fixed cost ($250,000). Class II and Class III mock-ups were estimated as a fraction of the structure cost of the first prototype. Trainers and miscellaneous other RDTE costs were treated as a throughput. Systems management and other data were estimated as a percentage of all other RDTE costs (8 percent).

Initial Investment

Flyaway costs represent the aggregate purchase price of all production aircraft to be built. Airframe costs were estimated by subsystem weight. Engine costs were estimated by power. Avionics, armament, photographic and miscellaneous equipment were throughputs. Attrition

rates were expressed in aircraft per 100,000 flight hours. Maintenance float was chosen from practice as 10 percent of the total fleet.

Initial spares were estimated as a percentage of flyaway costs by subsystem. Initial training and travel were based on an estimated table of organization and equipment (TOE) derived from the MMH/FH for the aircraft under study. Crew factors and maintenance and support factors, along with initial training and travel costs, were taken from the ARCSA II (Aviation Requirements for the Combat Structure of the Army) report published by the Army (Reference 13). Initial fuel stocks were based on the fuel flow in gallons per flight hour (90-day supply). Aircraft-related ground support equipment costs were estimated as 10 percent of flyaway, a factor obtained from the ARCSA II report. Nonaircraft supplies and equipment costs not directly associated with the aircraft were based on the ARCSA II dollars-per-head factor ($6,000).

Army Operation and Maintenance

Fuel costs - petroleum, oil, and lubricants (POL) - were based on the fuel flow in gallons per flight hour. Replacement parts and depot overhaul were estimated as a percentage of flyaway cost. Direct maintenance labor was based on the MMH/FH and the flying program of the operational aircraft. Pay and allowances (flight personnel) were based on ARCSA II rates applied to the flight personnel, as calculated in the model. Indirect maintenance costs also were based on ARCSA II rates and were applied to the portion of direct maintenance personnel performing nonmaintenance activity.

Replacement training and travel were based on ARCSA II factors applied to calculated TOE's as were support personnel pay and allowances, medical, and Army-wide costs.

Depot labor was based on depot MMH/FH and the prevailing depot maintenance labor rate and overhead costs.

The following cost ground rules were postulated for the aircraft study:

1.	Number of prototypes	-	10
2.	Production aircraft	-	2000
3.	Operational aircraft	-	1481
4.	Maintenance float	-	148
5.	Attrition aircraft	-	371
6.	Contractor flight test	-	1180 hr
7.	Customer flight test	-	4936 hr
8.	Fatigue test article	-	1
9.	Static test article	-	1
10.	Avionics	-	$25,000
11.	Peacetime flying hours	-	500/Year
12.	Years of operation	-	10
13.	Attrition rates	-	5/100,000 flight hr
14.	RDTE spares (percent of prototypes)	-	24
15.	Crew	-	2
16.	Dollars	-	1970
17.	Component test	-	All subsystems
18.	Systems management	-	Program

Maintenance man-hours at each level of support were developed in previous paragraphs. Additional data are summarized in Table XVII.

TABLE XVII. DATA FOR COST-EFFECTIVENESS STUDY OF UTILITY AIRCRAFT WITH REGENERATIVE AND NONREGENERATIVE ENGINES

	Regenerative Engines			Nonregenerative Engines	
	Effectiveness			Advanced-Technology	Available-Technology
	0.40	0.65	0.80		
Flyaway Cost/Aircraft -$ Million					
Airframe	.300	.296	.299	.297	.305
Engines	.037	.042	.065	.034	.036
Avionics	.025	.025	.025	.025	.025
Total	.362	.363	.389	.356	.366
Subsystem Spares - Percent of Flyaway Cost					
Initial Spares	41.0	41.0	41.0	41.0	43.0
Replenishment Spares, Depot Parts	85.3	85.3	85.3	83.8	88.9
Fuel Flow - Gal./hr	43.9	36.8	34.0	44.4	54.6

Life-Cycle Cost and Cost Effectiveness

The life-cycle costs of the five design aircraft have been calculated and are tabulated in Table XVIII. Considering the RDTE portion of the table, only the engine development costs differ markedly between the advanced-technology nonregenerative engine and the 0.65 effectiveness regenerative engine. Confining attention to these two engines in the data items listed under initial investment, flyaway costs and initial spares (which are a function of flyaway cost) were the only substantially different items. Table XVII illustrated that the differences in flyaway cost are caused solely by engine cost differences. Finally, in the list of items under operations and maintenance in Table XVIII, replenishment and depot parts and depot labor were quite different, and the fuel costs were much lower for the regenerative engine, of course. The total systems cost data illustrate that the greater costs associated with the development, production, and maintenance of the 0.65 effectiveness regenerative engine could not be overcome by the reduction in fuel costs. The same reasoning would hold true for the 0.40 effectiveness regenerative engine. For the engine with the 0.80 effectiveness regenerator, however, the aircraft gross weight is considerably higher than the other engines, and many cost items are larger as a consequence. The final results showed that the life-cycle costs were higher for the aircraft with regenerative engines, compared to the helicopter with the advanced-technology simple-cycle engine, but the difference was less than 1.0 percent for 0.40 and 0.65 effectiveness.

A frequency weighting methodology was developed to evaluate the cost-effectiveness of the regenerative-engine powered aircraft compared to a helicopter with simple-cycle engines. From data in the AIRCRAFT MISSION DEFINITION section of this report, Figures 8, 10, and 12, mission range requirements were determined for design and secondary missions at the 5-, 50-, and 95-percent frequency points. Cost-effectiveness parameters, in terms of ton-miles per dollar, were calculated for all the aircraft at these ranges, and cost-effectiveness ratios were calculated with the advanced-technology simple-cycle engine as the baseline. The results were averaged, weighting the extremes by a factor of 1.0 and the mid-range value by a factor of 4.0. After establishing a similar cost-effectiveness ratio for each mission, an overall cost-effectiveness for all missions was simply a weighted average of these average cost-effectiveness parameters for the

individual missions. The weights in this latter calculation were the frequencies of the individual missions in each level of conflict, from Table II.

Although the differences in life-cycle cost (comparing the aircraft with a simple-cycle engine and those with 0.40 and 0.65 effectiveness engines) were less than 1.0 percent, the differences in cost-effectiveness were greater. The aircraft with regenerative engines were less cost-effective than their counterparts with simple-cycle engines, and the difference became progressively greater with increasing recuperator effectiveness. Figure 60 demonstrates that by off-loading fuel, the aircraft with nonregenerative engines could carry greater payloads than those with regenerative engines for shorter distances than prescribed for the design mission. This result was obvious, since the regenerative engines burn less fuel per mile of operation and have less installed tankage. Applying these considerations to the cost-effectiveness studies, the short ranges for the design and secondary missions in Table XIX, corresponding to the different values of frequency of occurrence, had a major impact on the cost-effectiveness numbers.

These results suggested that increased range requirements for the utility helicopter might improve the relative life-cycle cost and cost-effectiveness of the regenerative engine-powered aircraft compared to those with simple-cycle engines.

Cost Sensitivity Analysis

The data in Table XVIII were analyzed to determine the impact on the aircraft life-cycle cost of engine costs and engine reliability and maintainability. Discussion of this analysis will be confined to the 0.65 effectiveness regenerative engine and the advanced-technology nonregenerative engine to illustrate typical trends.

Under RDTE, only a fraction of the recurring prototype costs (including prototype spares) and the engine development costs are dependent on engine cost predictions. The $8.5 million difference in RDTE is virtually all due to differences in engine development cost. Under initial investment, the engine portion of flyaway costs and initial spares (which are a percentage of flyaway costs) are dependent on engine cost predictions. The differences in production engine cost ($84 million compared with $68 million for 2000 engines,

TABLE XVIII. LIFE-CYCLE COSTS OF UTILITY HELICOPTERS WITH REGENERATIVE AND NONREGENERATIVE ENGINES - $ MILLION

	Regenerative Engines			Nonregen. Engines	
	Effectiveness			Advanced	Available
	0.40	0.65	0.80	Technology	Technology
RDTE					
Nonrecurring Engineering Design	4.389	4.352	4.383	4.371	4.480
Nonrecurring Tooling	9.841	9.747	9.886	9.781	10.127
Recurring Prototypes	20.030	19.960	20.550	19.860	20.410
Prototype Spares	4.807	4.790	4.932	4.766	4.898
Component Testing	14.552	14.552	14.552	14.552	14.552
Fatigue Test Article	1.966	1.959	1.975	1.965	1.969
Static Test Article	1.966	1.959	1.975	1.965	1.969
Flight Testing	60.661	60.661	60.661	60.661	60.661
Class I, II & III Mock-ups	2.500	2.492	2.511	2.498	2.500
Trainers & Misc. Other RDTE Costs	.579	.579	.579	.579	.579
System Management & Other Data	1.602	1.597	1.644	1.589	1.633
Engine Development	57.000	56.000	56.500	47.500	49.500
Total RDTE	179.893	178.648	180.148	170.087	173.278
Initial Investment					
Flyaway Costs	724.000	726.000	778.000	712.000	732.000

Flyaway Costs	724.000	726.000	778.000	712.000	732.000
Operational A/C	536.484	537.966	576.498	527.592	542.412
Maintenance Float	53.576	53.724	57.572	52.688	54.168
Attrition A/C	133.940	134.310	143.930	131.720	135.420
Initial Spares	57.538	60.344	74.312	55.122	58.509
Initial Training & Travel	239.495	239.495	239.499	239.426	239.699
Initial Fuel Stocks	.990	.833	.765	.990	1.238
Aircraft Related GSE	53.576	53.724	57.572	52.688	54.168
Nonaircraft	146.621	146.621	146.629	146.441	147.153
Total Initial Investment	1222.220	1227.017	1296.777	1206.667	1232.767
Army Operation and Maintenance					
Fuel (POL)	39.616	33.313	30.616	39.616	49.520
Replenishment & Depot Parts	149.054	154.530	183.861	143.008	151.225
Depot Labor	544.947	544.947	544.947	538.224	561.685
Direct Maintenance Labor	57.951	57.951	57.970	57.571	59.073
Pay & Allow. (Flight Personnel)	581.373	581.373	581.373	581.373	581.373
Indirect Maintenance Cost	18.341	18.341	18.346	18.220	18.696
Replacement Training & Travel	637.428	637.428	637.451	636.951	638.817
Support Personnel - Pay & Allow.	695.384	695.384	695.417	694.684	697.452
Medical & Army Wide Costs	366.552	366.552	366.573	366.102	367.881
Total Army O&M	3090.646	3089.819	3116.550	3075.756	3125.722
Total Systems Cost	4492.759	4495.484	4593.475	4482.510	4531.767
Total Systems Cost/Aircraft	3.034	3.035	3.102	3.006	3.060

TABLE XIX. RELATIVE COST-EFFECTIVENESS PARAMETERS FOR INDIVIDUAL MISSIONS AND OVERALL MISSION EFFECTIVENESS FOR UTILITY HELICOPTER WITH REGENERATIVE AND NONREGENERATIVE ENGINES

Intensity of Conflict	Low				Mid				High			
Mission Frequency - Percent												
Utility Mission	61				52				61			
Medical Evacuation Mission	28				44				22			
Observation Mission	11				4				17			
Total	100				100				100			
Percentile - Percent (a)	5	50	95	Average	5	50	95	Average	5	50	95	Average
Utility Mission												
Range - NM	3.	17.	43.		9.	14.	61.		25.	36.	73.	
Cost-Effectiveness Ratio												
Regenerative Engine - 0.40 Effectiveness	.988	.988	.988	.988	.985	.988	.988	.988	.985	.988	.987	.987
- 0.65 Effectiveness	.953	.955	.960	.956	.949	.952	.966	.954	.951	.961	.969	.961
- 0.80 Effectiveness	.917	.922	.930	.923	.916	.919	.935	.921	.916	.927	.938	.927
Advanced-Technology - Simple-Cycle Engine	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
Available-Technology- Simple-Cycle Engine	1.032	1.027	1.019	1.026	1.026	1.026	1.017	1.024	1.021	1.022	1.011	1.020
Medical Evacuation Mission												
Range - NM	13.	17.	53.		9.	13.	37.		45.	59.	80.	
Cost-Effectiveness Ratio												
Regenerative Engine - 0.40 Effectiveness	.988	.988	.988	.988	.985	.988	.988	.988	.991	.988	.987	.988
- 0.65 Effectiveness	.952	.955	.963	.956	.949	.952	.960	.953	.963	.966	.969	.966
- 0.80 Effectiveness	.919	.922	.932	.923	.916	.919	.927	.920	.929	.935	.941	.935
Advanced-Technology - Simple-Cycle Engine	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
Available-Technology- Simple-Cycle Engine	1.026	1.027	1.016	1.025	1.026	1.026	1.022	1.024	1.022	1.017	1.007	1.016
Observation Mission (b)												
Endurance - hr	.75	2.00	4.00		.50	1.00	1.60		.55	2.00	2.90	
Cost-Effectiveness Ratio												
Regenerative Engine - 0.40 Effectiveness				1.000				1.000				1.000
- 0.65 Effectiveness				1.000				1.000				1.000
- 0.80 Effectiveness				1.000				1.000				1.000
Advanced-Technology - Simple-Cycle Engine				1.000				1.000				1.000
Available-Technology- Simple-Cycle Engine				1.000				1.000				1.000
Overall Cost-Effectiveness Ratio												
Regenerative Engine - 0.40 Effectiveness				.989				.989				.989
- 0.65 Effectiveness				.961				.955				.969
- 0.80 Effectiveness				.931				.924				.941
Advanced-Technology - Simple-Cycle Engine				1.000				1.000				1.000
Available-Technology- Simple-Cycle Engine				1.023				1.023				1.016

(a) This is the percentage of occurrence of missions for which range or endurance is equal to or less than tabulated value.

(b) Aircraft effectiveness for utility and medical evacuation missions is defined in terms of the product of payload and range. However, payload and range are not applicable to the observation mission, which involves an observer and aircraft endurance, and so effectiveness and cost-effectiveness cannot be expressed in comparable terms. Since all the aircraft can perform the observation mission, the cost-effectiveness of all is defined as 1.0.

Table XVII) and in initial spares ($6.9 million compared with $5.3 million for the engine fraction of the spares) are responsible for most of the difference in the total initial investment. These are the only items in the calculation of aircraft life-cycle cost which are dependent on engine cost predictions.

Referring to Figure 58, the difference in rated power between the 0.65 effectiveness regenerative engine and the advanced-technology nonregenerative engine is only 4 percent. Since the engines are almost the same power class, it is evident that the regenerative-engine costs must be higher than those of the simple-cycle engine. The data in Table XVIII accurately reflect this predicted difference.

The predictions of engine reliability and maintainability data impact primarily on operation and maintenance cost items, although they also are reflected in initial training and travel under initial investment (which is calculated as a percentage of MMH/FH). The fraction of aircraft MMH/FH attributed to the engine was used to calculate that fraction of maintenance costs which are dependent upon the engine configuration. Costs of replenishment and depot parts are a varying percentage of flyaway costs - a higher percentage for the regenerative-engine aircraft because of the added complexity of the recuperator. Of these costs, $18.3 million was the portion attributed to the 0.65 effectiveness regenerative engine and $13.2 million attributed to the nonregenerative engine. Of the depot labor costs, $81.5 million was attributed to the regenerative engine and $74.8 million to the simple-cycle engine. Of the initial training and travel costs, under initial investment, $35.2 million was attributed to the regenerative engine and $33.3 million to the simple-cycle engine. Because the regenerative and nonregenerative engines are almost the same power class, the higher maintenance costs for the regenerative engine are certainly reasonable.

Summing the various engine-related costs that have been discussed, the total difference between those for regenerative and nonregenerative engines is $39.8 million, which is almost the same as the difference in life-cycle cost of the corresponding aircraft systems. Since the difference between the rated power of the engines is only 4 percent, the higher costs of the regenerative engine are understandable. Any change in the assumptions used to calculate engine cost, reliability, and maintainability data would only impact on the difference

in life-cycle costs, while the total life-cycle cost would still favor the nonregenerative engine in the utility aircraft.

Extended-Range Requirements

The range or endurance for the various mission roles which typified Army use of the utility helicopter was not sufficiently long to produce appreciable fuel savings with regenerative engines. Consequently, basic differences in development and production costs and maintenance requirements, for the regenerative engines compared to the nonregenerative engine, were too great to be offset by fuel savings. To investigate the impact of range on life-cycle cost and cost-effectiveness comparisons, a utility mission with a longer range requirement was postulated. The range for the design utility mission was extended from 150 NM to 300 NM. The trend relationships derived in the aircraft parametric studies were used.

The life-cycle costs of the aircraft with advanced-technology regenerative and nonregenerative engines, designed to perform a utility mission with a 300-NM range, are summarized in Table XXI. Even with this greater range requirement, the life-cycle costs were higher for the aircraft with regenerative engines, compared to the helicopter with an advanced-technology simple-cycle engine, but the differences among them were less than before. An interesting observation was that the 0.65 effectiveness regenerative engine offered nearly optimum life-cycle cost of the range of effectiveness values considered. The significant aircraft performance and weight parameters for the extended-range mission are listed in Table XX.

Vulnerability

The regenerative and nonregenerative advanced-technology engine designs in Reference 1 were only preliminary conceptual designs. Consequently, AiResearch was unable to develop realistic vulnerability data differentiating between the two concepts. Vulnerability parameters for the other aircraft subsystems would be virtually identical. Accordingly, no vulnerability analyses were conducted, but it seems very unlikely that differences in vulnerability between the regenerative and nonregenerative engines would have any impact on the final study results.

TABLE XX. SUMMARY OF SINGLE-ENGINE HELICOPTER CONFIGURATIONS PERFORMING DESIGN UTILITY MISSION, EXTENDED RANGE REQUIREMENTS (300-NM RANGE)

	Regenerative Engines			Advanced Techn. Simple-Cycle Engine
	Effectiveness			
	0.40	0.65	0.80	
Design Gross Weight, lb	8100.	7620.	7550.	7880.
Weight Empty, lb	5060.	4797.	4814.	4859.
Fixed Useful Load, lb (Incl. Mission Equipment)	780.	780.	780.	780.
Mission Fuel, lb	1060.	843.	756.	1049.
Payload, lb	1200.	1200.	1200.	1200.
Disc Loading, lb/ft^2	6.45	6.07	6.00	6.27
Installed Power, shp (Sea Level, 59°F, MRP)	1532.	1390.	1386.	1495.
Tail Rotor Power, shp	83.	76.	75.	79.
Total Equivalent Flatplate Area, F_e, ft^2	11.10	11.10	11.10	10.98
V_{CR}, KTAS	151.	149.	149.	148.

TABLE XXI. LIFE-CYCLE COSTS OF UTILITY HELICOPTERS WITH REGENERATIVE AND NON-REGENERATIVE ENGINES, EXTENDED RANGE REQUIREMENTS - $ MILLION

	Regenerative Engines			Simple-Cycle Engine
	Effectiveness			
	0.40	0.65	0.80	
RDTE				
Nonrecurring Engineering Design	4.8	4.7	4.7	4.7
Nonrecurring Tooling	10.9	10.6	10.6	10.7
Recurring Prototypes	22.0	21.5	22.0	21.6
Prototype Spares	5.3	5.2	5.3	5.2
Component Testing	14.6	14.6	14.6	14.6
Fatigue Test Article	2.0	2.0	2.0	2.0
Static Test Article	2.0	2.0	2.0	2.0
Flight Testing	60.7	60.7	60.7	60.7
Class I, II, & III Mock-ups	2.6	2.5	2.6	2.6
Trainers & Misc. Other RDTE Costs	.6	.6	.6	.6
System Management & Other Data	10.0	9.9	10.0	9.9
Engine Development	64.0	61.0	61.0	52.5
Total RDTE	199.5	195.3	196.1	187.1
Initial Investment				
Flyaway Costs	816.0	798.0	848.0	792.0

Initial Investment				
Flyaway Costs	816.0	798.0	848.0	792.0
Operational A/C	604.3	590.9	627.9	586.5
Maintenance Float	60.4	59.1	62.8	58.7
Attrition A/C	151.3	148.0	157.3	146.8
Initial Spares	67.5	68.1	83.3	63.2
Initial Training & Travel	239.5	239.5	239.5	239.4
Initial Fuel Stocks	1.4	1.1	1.0	1.3
Aircraft Related GSE	60.4	59.1	62.8	58.7
Nonaircratt	146.6	146.6	146.6	146.4
Total Initial Investment	1,331.4	1,312.4	1,381.2	1,301.0
Army Operation and Maintenance				
Fuel (POL)	54.9	43.2	38.7	53.1
Replenishment & Depot Parts	173.0	173.1	204.7	162.7
Depot Labor	544.9	544.9	544.9	538.2
Direct Maintenance Labor	58.0	58.0	58.0	57.6
Pay & Allow. (Flight Personnel)	581.4	581.4	581.4	581.4
Indirect Maintenance Cost	18.3	18.3	18.3	18.2
Replacement Training & Travel	637.4	637.4	637.5	637.0
Support Personnel - Pay & Allow.	695.4	695.4	695.4	694.7
Medical & Army-Wide Costs	366.6	366.6	366.6	366.1
Total Army O&M	3,129.9	3,118.3	3,145.5	3,109.0
TOTAL SYSTEMS COST	4,660.8	4,626.0	4,722.8	4,597.1
Total Systems Cost/A/C	3.147	3.124	3.189	3.104

CONCLUSIONS

This report completes the conceptual design study of aircraft powered by regenerative and nonregenerative engines performed by The Boeing Company under Contract DAAJ02-70-C-0061, Regenerative Engine Powered Aircraft Design Study. Boeing selected a design utility mission and secondary medical evacuation and observation missions which typified Army use of the utility helicopter in all intensities of conflict. Reference 1 provided advanced engine and recuperator data for the lightweight regenerative engine designs of approximately 1000 shp used in the study. Component performance data consistent with the reference report were used to develop performance for a nonregenerative engine, but the compressor pressure ratio used was more nearly optimum for simple-cycle engines. The following principal conclusions have resulted from the study:

1. The engine designs in Reference 1 were a lightweight, integrated configuration utilizing an annular tube-type recuperator wrapped around the turbomachinery. The recuperator served as the structural backbone of the engine assembly. Conversely, regenerative engines tested to date have been derived from existing turboshaft engines, modified to accommodate a bolt-on recuperator.

   Because of the advanced level of the engine performance parameters, however, the improvements in SFC which could be achieved with the regenerative engines, compared with the optimum advanced-technology simple-cycle engine, were relatively small (15 to 17 percent). Coupled with the missions predicated for the utility helicopter, the small improvements in SFC meant that the fuel savings in the aircraft installation were almost equal to the increased weight of the regenerative engine.

2. The lightest aircraft concept, from the viewpoint of both empty weight and gross weight, had a 0.65 effectiveness recuperative engine. However, the weight of this aircraft was only two percent less than that of the aircraft powered by an advanced-technology simple-cycle engine.

3. The aircraft with the advanced-technology simple-cycle engine had the lowest life-cycle cost and the optimum cost-effectiveness. Higher development and production costs and maintenance requirements for the regenerative engines were too great to be offset by savings in fuel. Again the differences among the various aircraft were very small (within 1 percent on life-cycle cost). Although a longer-range mission was also investigated for the helicopter, the trends in life-cycle cost still favored the nonregenerative engine.

4. Variable power-turbine stator vanes, which provide variable flow characteristics for the turbine, would permit part-power operation at constant high turbine-inlet temperatures, and would realize the maximum benefit in engine SFC over the entire operating range of the engine. Compared with a regenerative engine with fixed turbine geometry, the regenerative engine with variable power-turbine geometry would not offer much improvement in SFC at relatively high power settings, but at 60 percent power it would have a 6 percent lower SFC. This in turn is 21 percent less than the SFC of the advanced-technology simple-cycle engine. Comparing the fixed-geometry and variable-geometry regenerative engines, however, the fuel savings at the high power settings typical of the helicopter flying a utility mission would be negligible, and the added weight and complexity of variable-turbine designs would not be warranted.

Potentially, the regenerative-engine powered aircraft would be superior for longer range and endurance requirements, particularly if considerable loiter time were involved. In the latter case, variable power-turbine geometry could offer further performance benefits. However, other aspects of the regenerative-engine installation such as reduced infrared signature, improved vulnerability characteristics, and reduced noise, could be exploited, but were not included within the scope of this study.

LITERATURE CITED

1. McDonald, Colin F., STUDY OF A LIGHTWEIGHT INTEGRAL REGENERATIVE GAS TURBINE FOR HIGH PERFORMANCE, AiResearch Manufacturing Company, The Garrett Corporation; USAAVLABS Technical Report 70-39, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, August 1970, AD 877 464.

2. ANNEX D - UTTAS AND LTTAS SCENARIOS (U), Action Control Control Number 10705, United States Army Combat Developments Command, May 1968 (C).

3. LOW INTENSITY SCENARIOS - SECTION III OF ANNEX D - UTTAS STUDY (U), Action Control Number 10705, United States Army Combat Developments Command, April 1968 (C).

4. MID INTENSITY SCENARIOS - UTTAS STUDY (U), Action Control Number 10705, United States Army Combat Developments Command, April 1968 (C).

5. HIGH INTENSITY SCENARIOS - UTTAS STUDY (U), Action Control Number 10705, United States Army Combat Developments Command, April 1968 (C).

6. HEAT REGENERATIVE SYSTEM FOR T53 SHAFT TURBINE ENGINES, Lycoming Division, AVCO Corporation; USAAVLABS Technical Report 65-37, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, July 1965, AD 620 245.

7. Wheeler, A. J., H. R. Dolf, V. J. Klein, and J. Acurio, SMALL GAS TURBINE ENGINE COMPONENT TECHNOLOGY REGENERATOR RESEARCH, The Boeing Company, Turbine Division; USAAVLABS Technical Report 66-90, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, January 1967, AD 809 557L.

8. Privoznik, Edward J., T63 REGENERATIVE ENGINE PROGRAM, Allison Division, General Motors Corporation; USAAVLABS Technical Report 68-9, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, May 1968, AD 675 444.

9. Curbishley, G., W. Larson, D. W. McGrath, E. W. Gellerson, HOT CORROSION RESISTANCE OF MATERIALS FOR SMALL GAS TURBINE RECUPERATORS, The Garrett Corporation, AiResearch Manufacturing Company; USAAVLABS Technical Report 69-92, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, December 1969, AD 866 981.

10. AERONAUTICAL PROPULSION EXPLORATORY DEVELOPMENT: INDUSTRY BRIEFING, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, 8 September 1969, p. 112.

11. Hoerner, S. F., FLUID DYNAMIC DRAG, Midland, New Jersey, S. F. Hoerner Publisher, 1965.

12. PRELIMINARY DESIGN STUDY - UTILITY TACTICAL TRANSPORT AIRCRAFT SYSTEM (UTTAS) (U), The Boeing Company, Vertol Division, Philadelphia, Pennsylvania, Document D8-2195-1, 14 November 1968 (C).

13. Bennett, W., SYSTEMS COST ANALYSIS FOR AVIATION REQUIREMENTS FOR THE COMBAT STRUCTURE OF THE ARMY II (ARCSA II) (U), Planning Research Corporation, Washington, D.C., PRC R-894, 31 March 1967, AD 380 509 (C).

14. Messerlie, R. L., and D. M. Cox, STUDY OF VARIABLE TURBINE GEOMETRY FOR SMALL GAS TURBINE ENGINES, Allison Division, General Motors Corporation; USAAVLABS Technical Report 67-19, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, April 1967, AD 653 430.

APPENDIX I

TWIN-ENGINE AIRCRAFT CONFIGURATION STUDIES

Twin engines in the utility transport aircraft would enhance mission reliability and offer a desirable asset for the helicopter operating in a battlefield environment. Alternative aircraft conceptual designs included twin-engine aircraft for the same mission and payload. With the exception of increased mission reliability, however, all other factors favored single-engine installations compared with twin engines. Analyses and data pertaining to twin-engine aircraft using regenerative and nonregenerative engines of approximately 500 shp have been assembled in this Appendix.

TURBOSHAFT ENGINE DESIGN PARAMETERS

Performance and weight characteristics have been developed for simple-cycle turboshaft engines utilizing advanced technologies and for simple-cycle engines utilizing available technologies, nominally sized at 500 shp.

Advanced-Technology Engine

Reference 1 provided design-point performance data for advanced-technology regenerative and nonregenerative engines of approximately 1000 shp, all at 9:1 compressor pressure ratio. A parametric study of the effect of pressure ratio on design-point performance led to the selection of 14:1 as the optimum value for the 1000-shp advanced-technology turboshaft engine, and this 14:1 pressure ratio was also the design point used for the 500-shp advanced-technology simple-cycle engine.

High turbine-inlet temperatures would pose a more serious problem in the smaller engine - the size of the turbine blades and vanes would cause difficulty in fabricating the required internal cooling-air passages and film-cooling slots. Consequently, the design turbine-inlet temperature was limited to 2250°F in the advanced-technology 500-shp engine, to allow for these considerations.

Due to the impact of clearances, material thicknesses, and tolerances, the smaller size components could not achieve the design-point performance of those in the 1000-shp engine. The performance characteristics of the 500-shp advanced-technology engine were obtained by downgrading the component performance of the larger engine, and the design-point data are summarized in the first column of Table XXII. The changes in component data, compared with the 1000-shp advanced-technology engine, are as follows (expressed in decimal fractions):

TABLE XXII. 500-SHP SIMPLE-CYCLE TURBOSHAFT ENGINE DESIGN-POINT PARAMETERS

Parameters	Advanced-Technology	Available-Technology
Compressor		
Inlet Airflow, lb/sec	2.84	3.36
Pressure Ratio	14.0	10.0
Adiabatic Efficiency*	.788	.78
Exit Temperature, °F	778.	664.
Cooling-Air Bleed/Inlet Airflow*	.045	.057
Leakage/Inlet Airflow*	.040	.023
Combustor		
Efficiency	.98	.985
Fuel/Compressor Inlet Airflow	.0225	.0224
Pressure Loss*	.04	.05
Gas Generator Turbine		
Inlet Temperature, °F	2250.	2150.
Inlet Flow, lb/sec	2.663	3.166
Mechanical Efficiency*	.975	.99
Exit Temperature, °F	1597.	1608.
Adiabatic Efficiency*	.87	.85
Pressure Ratio	4.00	3.30
Interstage Turbine Diffuser		
Pressure Loss*	.02	-
Temperature, °F (Cooling Air Mixed)	1587.	1554.
Power Turbine		
Inlet Temperature, °F	1587.	1554.
Inlet Flow, lb/sec	2.792	3.358
Exit Temperature, °F	1136.	1179.
Adiabatic Efficiency*	.89	.86
Pressure Ratio	3.20	2.72
Exhaust Diffuser		
Pressure Ratio	1.03	1.058
Specific Power, hp/lb/sec	176.0	149.0
Shaft Power, hp	500.	500.
SFC, lb/hr/hp	.465	.540

*Efficiencies, pressure losses, bleed and leakage flows expressed as decimal fractions.

1. Compressor efficiency decreased .02

2. Cooling air and leakage flow increased .01

3. Combustor efficiency decreased .01

4. Combustor pressure loss increased .01

5. Gas-generator turbine efficiency, compared to the trend data, decreased .01

6. Power turbine efficiency, compared to the trend data, decreased .01

The resulting specific power and SFC were not as good as the corresponding parameters for the 1000-shp advanced-technology engine, but the performance projection was consistent with data trends to smaller engines.

Off-design performance was developed for the advanced-technology simple-cycle engine, to be used in performance calculations for the twin-engine aircraft. The compressor efficiency trend for the 1000-shp engine also was used for the 500-shp engine, with the level of values decreased .02 throughout the operating range to match the assumed design point. The efficiency characteristic in Figure 61 was used for power-turbine performance at optimum output-shaft speed. The results of the off-design performance calculation are presented in Figure 62, showing referred shaft horsepower and referred fuel flow plotted as a function of referred turbine-inlet temperature for various forward flight speeds (at optimum output shaft speed). The corrections for nonoptimum output-shaft speed were the same as those previously presented in Figures 20 and 21 of the main text of this report. The turbine-inlet temperatures selected were 2125°F for Normal Rated Power and 2250°F for Military Rated Power.

The dry weight of the 500-shp advanced-technology simple-cycle engine was estimated from general trends for small turboshaft engines as a function of rated shaft horsepower, using the weight of the 1000-shp engine as a starting point. The dry weight of the simple-cycle engine has been included in the correlation of weights for 500-shp advanced technology regenerative and nonregenerative engines in Figure 68.

Available-Technology Engine

Component data correlations were generated for small turboshaft engines under development or proposed for development utilizing available technologies, and compressor and turbine design-point efficiency trends were previously shown as Figures 25 and 26 in the main text of this report. The

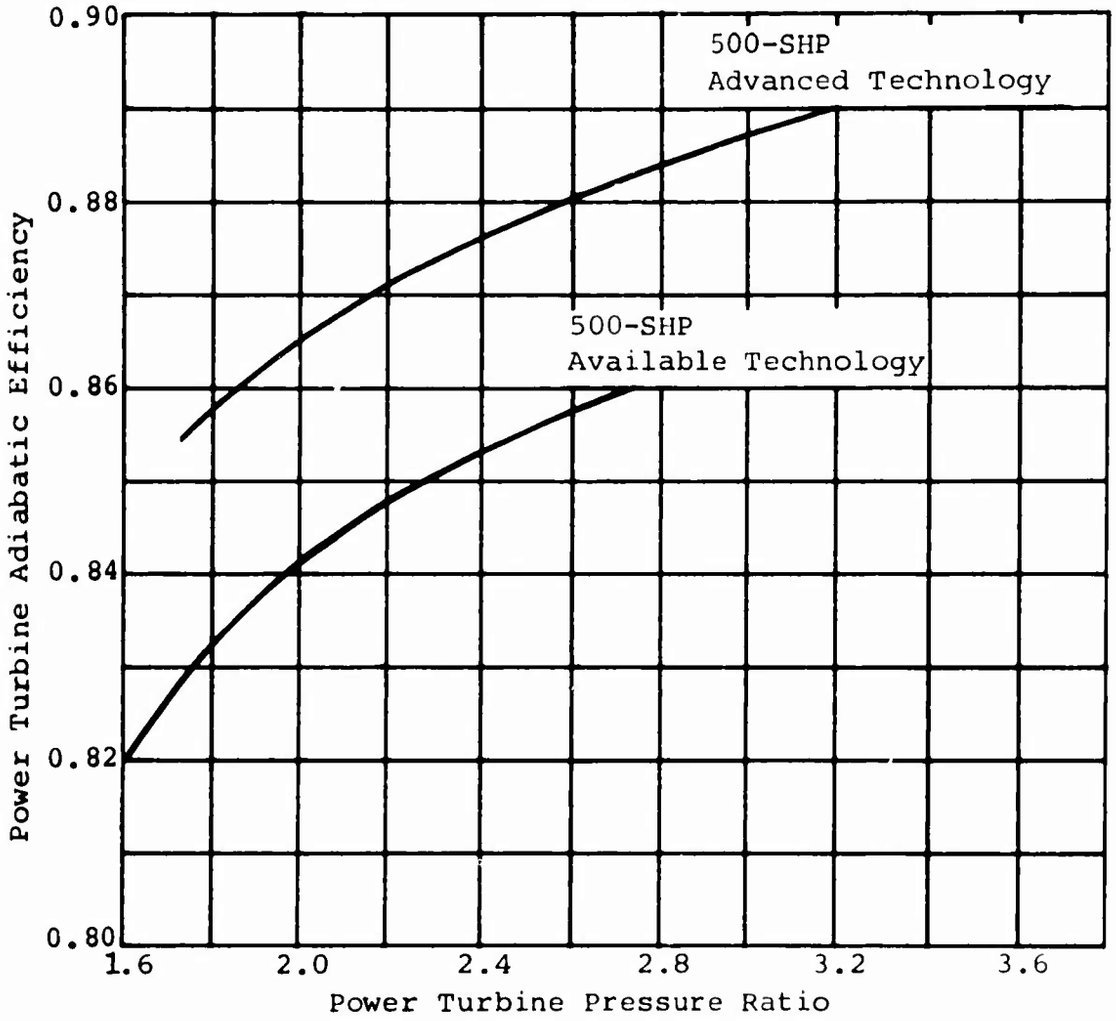


Figure 61. Power Turbine Off-Design Efficiency Trend for Nominal 500-SHP Simple-Cycle Engines.

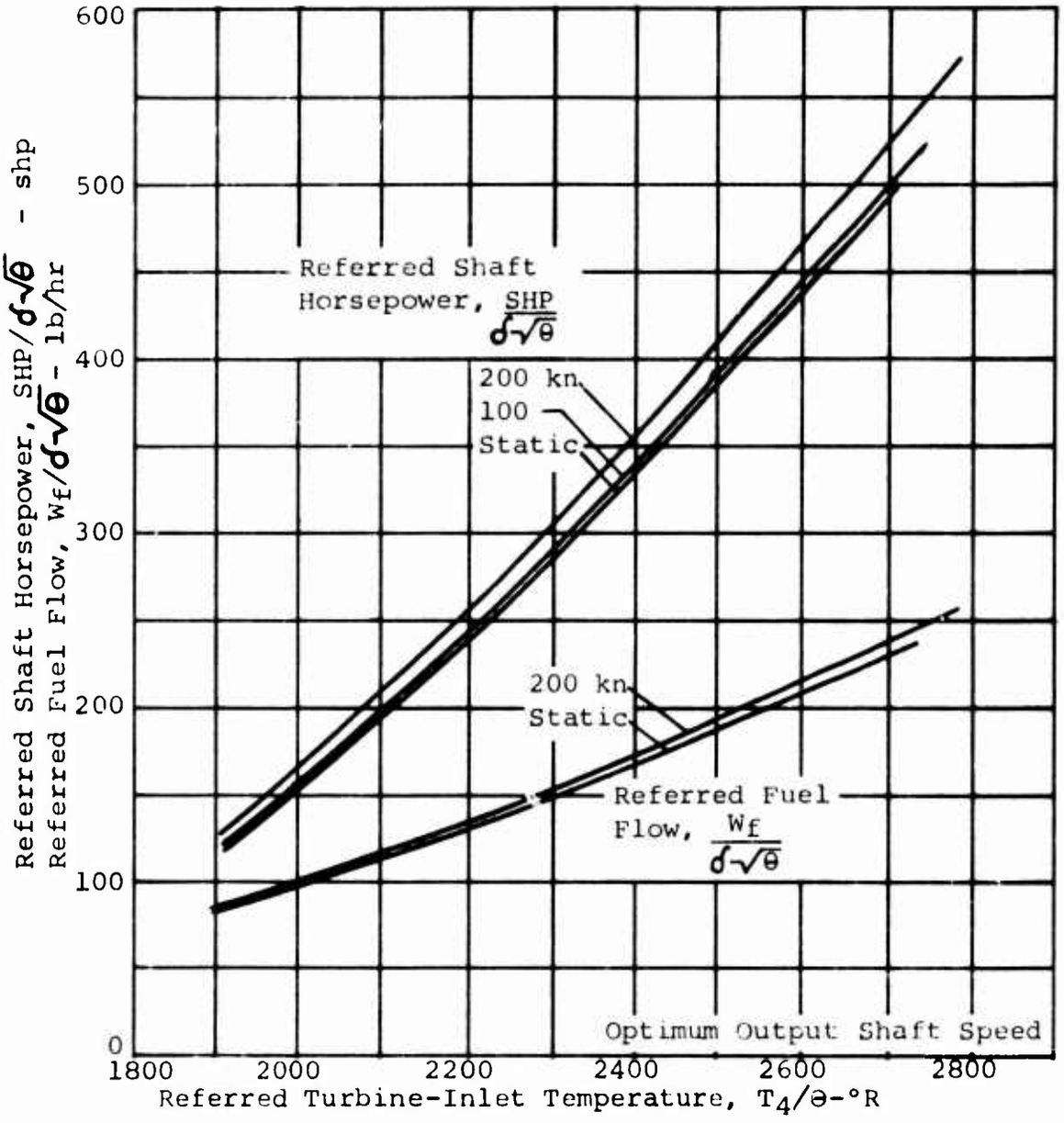


Figure 62. 500-SHP Advanced-Technology Simple-Cycle Engine Performance.

component trend data were used to define design-point component performance for the 500-shp simple-cycle engine, and to calculate overall performance, which was consistent with trends of specific power and SFC (second column of Table XXII). The selection of a 10:1 compressor design-point pressure ratio was in accord with the trends for engines in this power class. The turbine-inlet temperature of 2150°F was somewhat less than that for the 1000-shp engine, for the same reasons expressed in the discussion of the advanced-technology engine. Compressor and turbine efficiencies were selected from the trend curves. Percentages of cooling-air and leakage flows were kept the same as in the larger engine, despite the lower pressure ratio and turbine-inlet temperature. The resulting specific power and SFC from Table XXII were in agreement with the overall performance trends, and the data in the table were used for the 500-shp available-technology engine in subsequent study tasks.

The compressor efficiency trend in Figure 63 was used to calculate off-design performance for the available-technology engine. The power-turbine efficiency trend was plotted in Figure 61. The resulting curves of referred power and fuel flow were plotted in Figure 64. A turbine-inlet temperature of 2025°F was selected for Normal Rated Power and 2150°F for Military Rated Power.

From weight trends for available-technology engines, a value of 2.85 was selected for the power-to-weight ratio of the 500-shp engine.

REGENERATIVE ENGINE DESIGN PARAMETERS

Performance and weight characteristics have been developed for three 500-shp regenerative engines, with recuperator effectiveness of 0.40, 0.65, and 0.80 and recuperator pressure losses of 10.0, 6.0, and 4.0 percent, respectively. The design-point compressor pressure ratio was 9:1, as with the 1000-shp engines in Reference 1. The 2300°F design turbine-inlet temperature of the 1000-shp engines would be much more difficult to attain in the smaller engines. As with the simple-cycle configurations, the design turbine-inlet temperature was limited to 2250°F in the 500-shp regenerative engines.

Size effects in the smaller components would prevent their achieving the performance of those in the 1000-shp engines. Therefore, the performance characteristics of the 500-shp regenerative engines were obtained in the same manner as the simple-cycle arrangements. Design point data are summarized in Table XXIII. The specific power and SFC resulting from the component performance changes are also presented in this table.

TABLE XXIII. 500-SHP ADVANCED-TECHNOLOGY REGENERATIVE ENGINE DESIGN-POINT PARAMETERS

	Recuperator Effectiveness		
	.40	.65	.80
Compressor			
Inlet Airflow, lb/sec	3.08	2.98	2.93
Pressure Ratio	9.0	9.0	9.0
Adiabatic Efficiency*	.80	.80	.80
Exit Temperature, °F	614.	614.	614.
Cooling-Air Bleed/Inlet Airflow*	.04	.04	.04
Leakage/Inlet Airflow*	.03	.03	.03
Recuperator - Air-Side			
Inlet Flow, lb/sec	2.864	2.771	2.725
Pressure Loss*	.04	.024	.016
Exit Temperature, °F	904.	1069.	1164.
Combustor			
Efficiency*	.98	.98	.98
Fuel/Compressor Inlet Airflow	.0210	.0186	.0172
Pressure Loss*	.04	.04	.04
Gas Generator Turbine			
Inlet Temperature, °F	2250.	2250.	2250.
Inlet Flow, lb/sec	2.929	2.826	2.775
Mechanical Efficiency*	.975	.975	.975
Exit Temperature, °F	1755.	1754.	1753.
Adiabatic Efficiency*	.865	.865	.865
Pressure Ratio	2.77	2.78	2.79
Interstage Turbine Diffuser			
Pressure Loss*	.03	.03	.03
Temperature, °F (Cooling-Air Mixed)	1727.	1725.	1725.
Power Turbine			
Inlet Temperature, °F	1727.	1725.	1725.
Inlet Flow, lb/sec	3.052	2.945	2.892
Exit Temperature, °F	1320.	1303.	1296.
Adiabatic Efficiency*	.90	.90	.90
Pressure Ratio	2.62	2.72	2.78
Exhaust Diffuser			
Pressure Ratio	1.04	1.04	1.04
Recuperator - Gas-Side			
Inlet Flow, lb/sec	3.052	2.945	2.892
Pressure Loss*	.06	.036	.024
Exit Temperature, °F	1068.	901.	802.
Specific Power, shp lb sec	162.6	168.0	170.5
Shaft Power, shp	500.	500.	500.
SFC, lb/hr hp	.466	.398	.362

*Efficiencies, effectiveness, pressure losses, bleed and leakage flows expressed as decimal fractions.

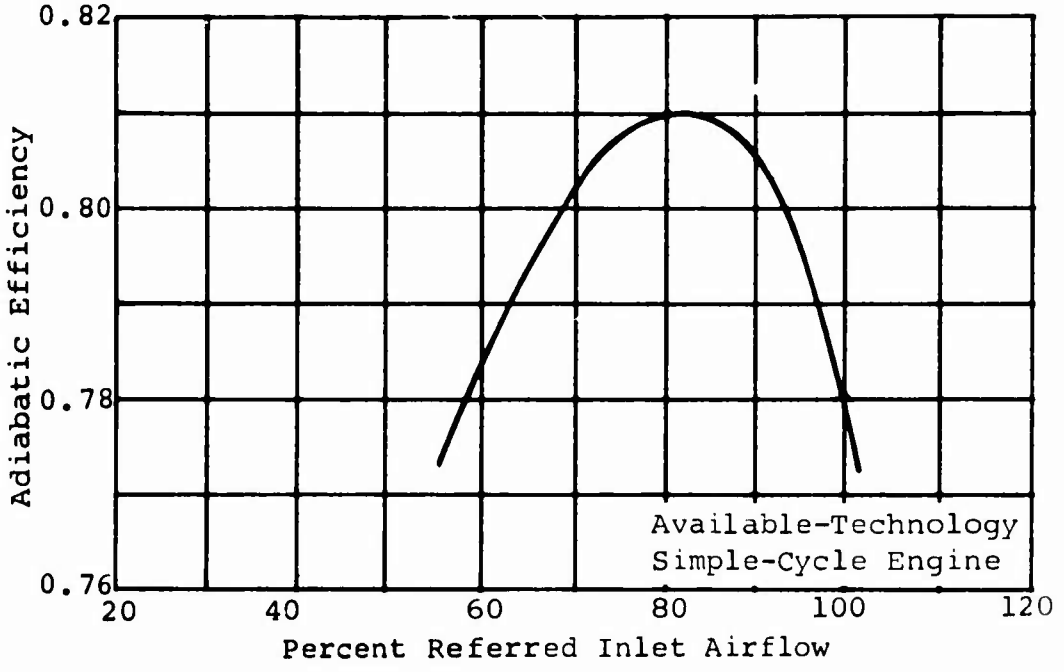


Figure 63. Compressor Efficiency Characteristic Along Engine Operating Line for 500-SHP Available-Technology Engine.

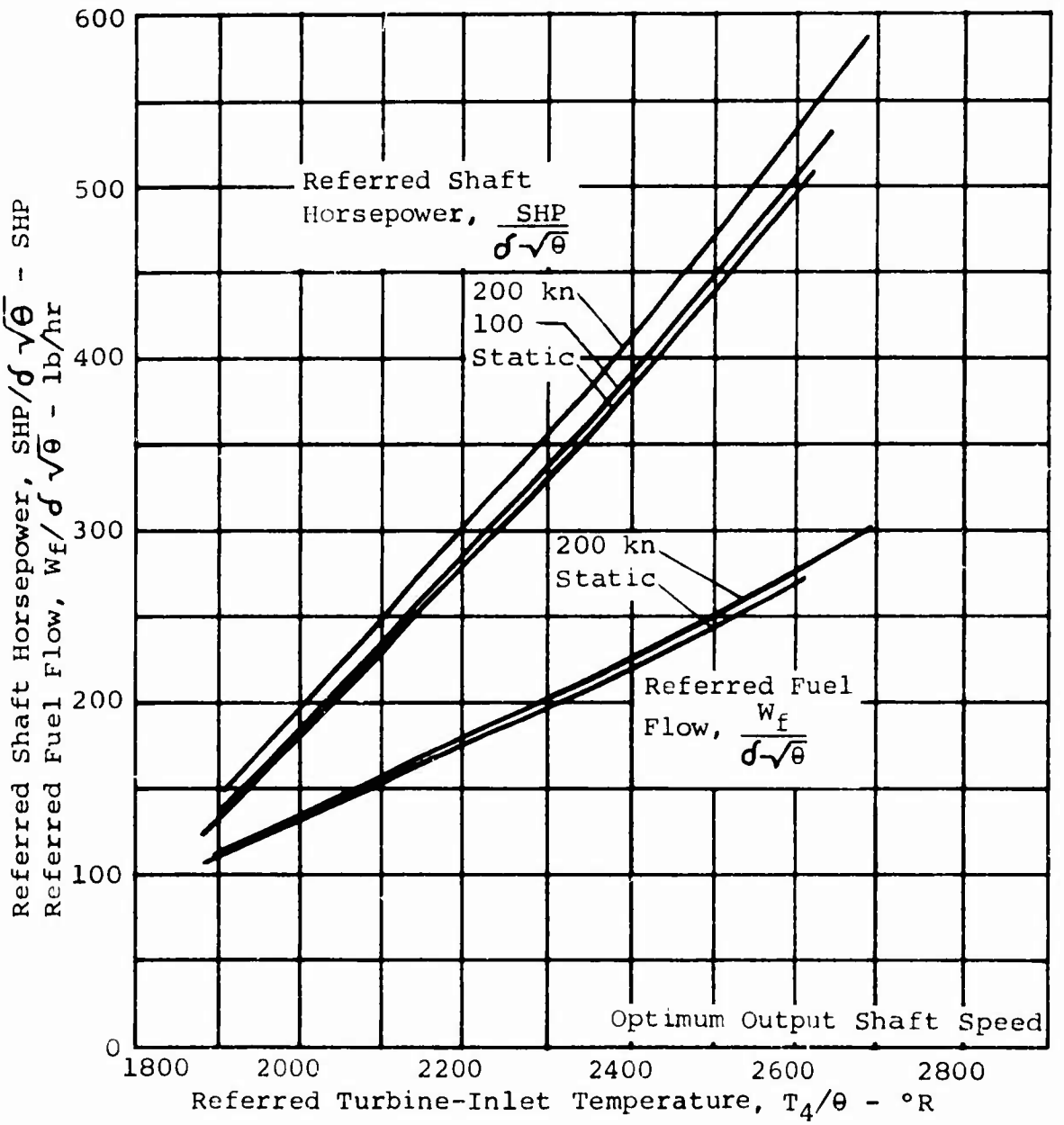


Figure 64. 500-SHP Available-Technology Simple-Cycle Engine Performance.

Off-design performance data were calculated for the three regenerative engines. The compressor efficiency trend developed for the 1000-shp engines was also used for the 500-shp engines, with the efficiencies decreased .02 throughout the operating range to match the assumed design-point. Similarly, the power turbine efficiency characteristic for the 1000-shp engines was used for the 500-shp engines, with the level of efficiencies downgraded to match the design-point value. The changes in recuperator performance at part-power settings (increased effectiveness and decreased pressure loss) were previously plotted in Figures 36 and 37 in the main text of this report. The resulting referred power and fuel flow for the three 500-shp regenerative engines were plotted in Figures 65, 66, and 67.

The weights assumed for the advanced-technology 500-shp regenerative and nonregenerative engines are shown in Figure 68. These data were developed from trends of weight as a function of shaft horsepower for small turboshaft engines. The increasing slope of the weight curve as a function of recuperator effectiveness is a result of the increased weight of the recuperator. This engine weight increase more than offset the decreased fuel requirements resulting from improvements in SFC at high values of effectiveness. Consequently, the optimum recuperator from an aircraft gross weight standpoint had an effectiveness considerably less than 0.80.

TWIN-ENGINE CONCEPTUAL DESIGNS

Twin-engine aircraft were designed for the utility mission and payload defined in the AIRCRAFT MISSION DEFINITION section of this report. That section also defined the OEI capability based upon Army requirements for multiengine aircraft, which established that the helicopter must have go-home capability with one engine inoperative. The installed power calculated for the twin-engine configurations had to meet or exceed both hover and OEI power requirements.

Conceptual aircraft designs were produced for four alternative twin-engine helicopters, with the following propulsion systems:

1. Internally-mounted regenerative engines
2. Internally-mounted nonregenerative engines
3. Externally-mounted regenerative engines
4. Externally-mounted nonregenerative engines

Data for the aircraft with internally-mounted engines are assembled in Table XXIV, while Table XXV shows aircraft with externally-mounted engines.

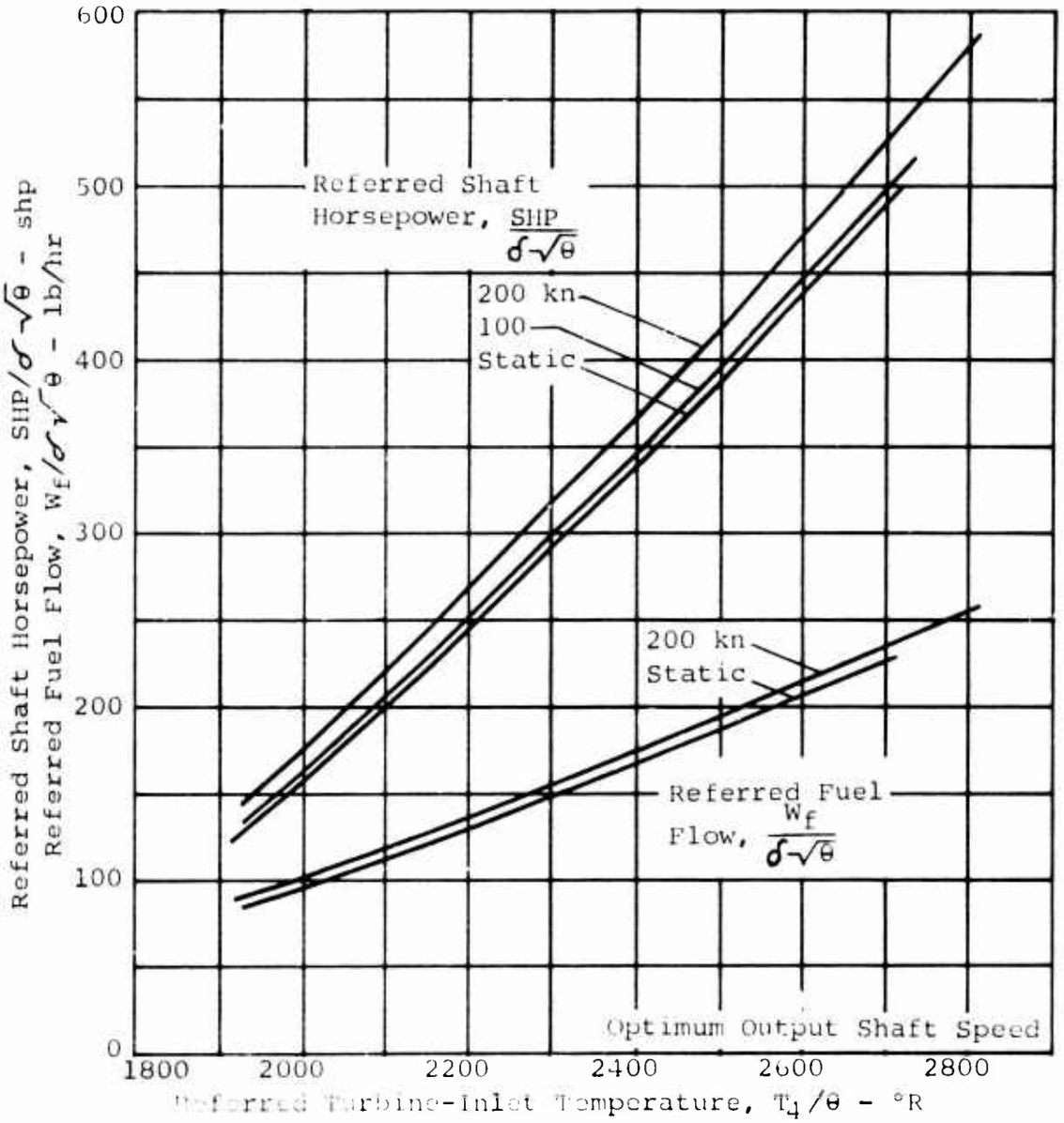


Figure 65. 500-SHP Advanced-Technology Regenerative Engine Performance (Design-Point Effectiveness = 0.40).

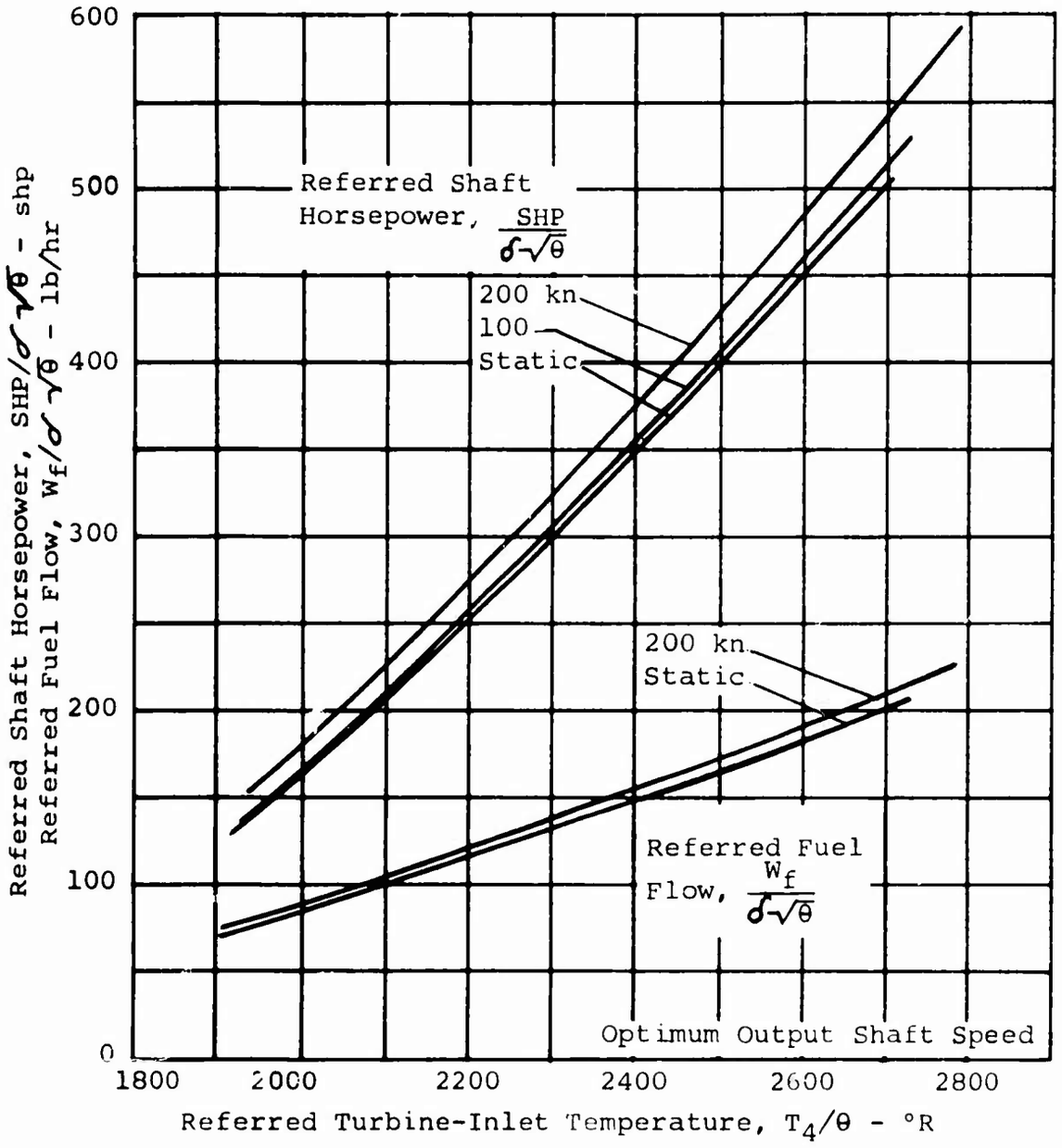


Figure 66. 500-SHP Advanced-Technology Regenerative Engine Performance (Design-Point Effectiveness = 0.65).

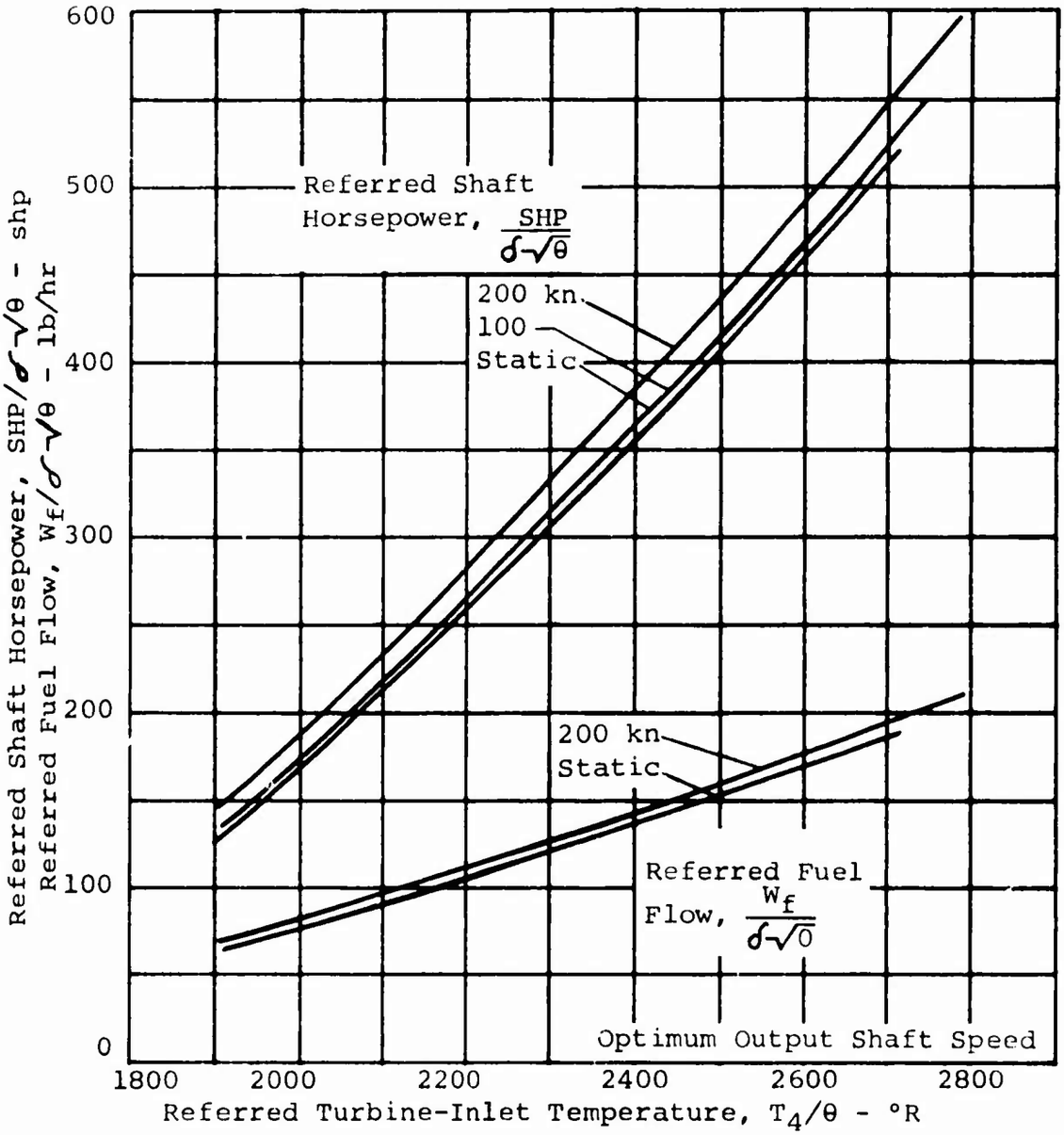


Figure 67. 500-SHP Advanced-Technology Regenerative Engine Performance (Design-Point Effectiveness = 0.80).

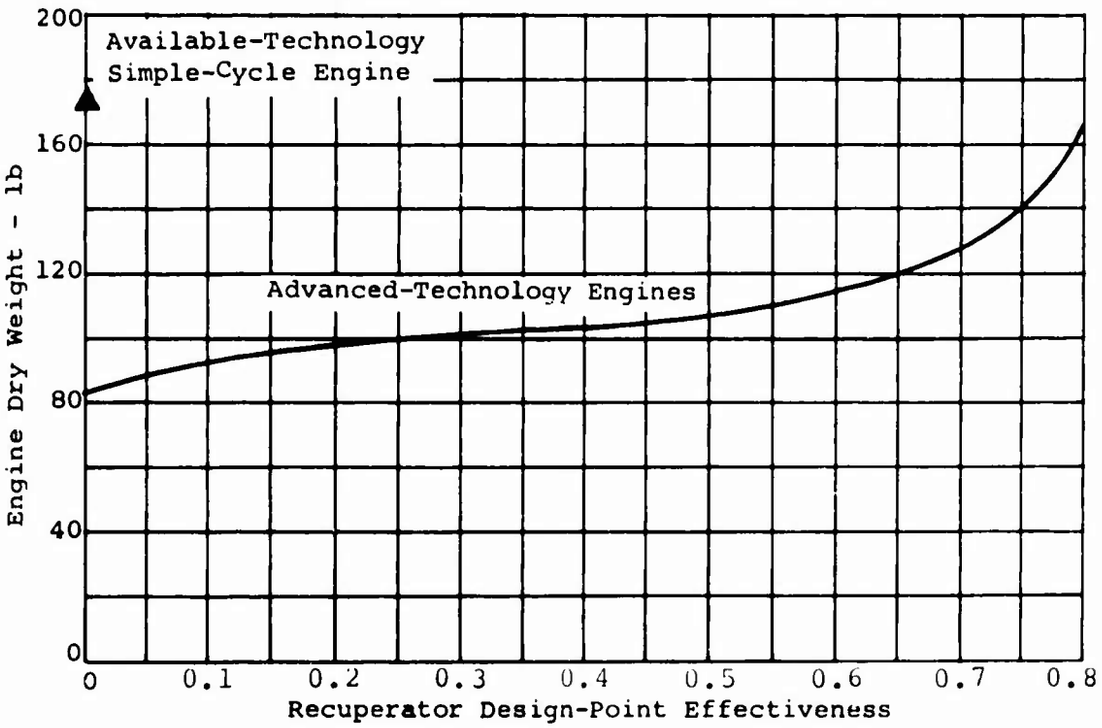


Figure 68. Dry Weights for 500-SHP Regenerative and Non-regenerative Engines.

TABLE XXIV. SUMMARY OF TWIN-ENGINE HELICOPTER CONFIGURATIONS WITH INTERNALLY-MOUNTED ENGINES

	Regenerative Engines			Non-regenerative Engines	
	Effectiveness			Adv.	Avail.
	0.40	0.65	0.80	Techn.	Techn.
Design Gross Weight, lb	7278	6909	7146	7029	7665
Weight Empty, lb	4665	4472	4677	4474	4947
Fixed Useful Load, lb (Incl.Mission Equipment)	780	780	780	780	780
Mission Fuel, lb	623	447	479	565	728
Payload, lb	1200	1200	1200	1200	1200
Disc Loading, lb/ft^2	5.78	5.50	5.68	5.60	6.08
Installed Power, shp (Sea Level, 59^oF, MRP)	1314	1228	1275	1233	1380
Rated Power/Engine, shp (Sea Level, 59^oF, MRP)	657.	614.	638.	617.	690.
Transmission Rating, shp (4000 ft, 95^oF, MRP)	972	910	942	923	1039
Tail Rotor Power, shp	76.3	71.0	74.1	72.3	71.0
Total Equivalent Flat-Plate Area, F_e, ft^2	12.32	12.32	12.32	12.22	12.22
V_{CRUISE}, TAS	144.	142.5	143.6	144.	145.
V_{OPT}, TAS (5000 ft, 95^oF)	74.	73.	73.5	74.	75.
Rate of Climb at V_{OPT}, fpm (5000 ft, 95^oF, OEI, MRP)	304.	100.	271.	100.	181.

TABLE XXV. SUMMARY OF TWIN-ENGINE HELICOPTER CONFIGURATIONS WITH EXTERNALLY-MOUNTED ENGINES

	Regenerative Engines			Non-regenerative Engines	
	Effectiveness			Adv. Techn.	Avail. Techn.
	0.40	0.65	0.80		
Design Gross Weight, lb	7248	6878	7100	7055	7654
Weight Empty, lb	4626	4445	4628	4507	4962
Fixed Useful Load, lb (Incl.Mission Equipment)	780	780	780	780	780
Mission Fuel, lb	632	443	482	558	702
Payload, lb	1200	1200	1200	1200	1200
Disc Loading, lb/ft^2	5.77	5.47	5.65	5.62	6.08
Installed Power, shp (Sea Level, 59^{o}F, MRP)	1305	1219	1263	1248	1398
Rated Power/Engine, shp (Sea Level, 59^{o}F, MRP)	653	610	632	624	699
Transmission Rating, shp (4000 ft, 95^{o}F, MRP)	966	903	936	934	1052
Tail Rotor Power, shp	75.0	70.4	73.1	73.0	82.2
Total Equivalent Flat-Plate Area, F_e, ft^2	16.27	16.27	16.27	14.75	14.75
V_{CRUISE}, TAS	145.	140.5	144.	149.	151.7
V_{OPT}, TAS (5000 ft, 95^{o}F)	74.	72.	73.	74.	76.
Rate of Climb at V_{OPT}, fpm (5000 ft, 95^{o}F, OEI, MRP)	213.	>100	188.	>100	316.

A drag buildup was calculated for each of the twin-engine aircraft, and total equivalent flat-plate area is included in the above tables. The flat-plate area for the aircraft with externally-mounted engines was significantly larger due to the induced drag of the stub-wing appendages used for engine mounting.

The aerodynamic and performance limits and constants used in the single-engine aircraft performance calculations were the same for these twin-engine aircraft analyses, including:

- Four-bladed rotor
- Rotor diameter = 40 ft
- Rotor solidity limit, $\sigma_{LIMIT} = 0.07$
- Two-bladed tail rotor
- Tail-rotor solidity = 0.11
- Tail-rotor efficiency = 0.712
- Transmission efficiency = 0.97

Figures 45, 47, and 49, presented earlier in the main text, showed the performance data used in these analyses, and Figure 46 presented the fuselage download.

The twin-engine concepts with externally-mounted engines would be subject to greater download due to the wing-like external mounting structure, and this additional increment is plotted in Figure 69. The requirement for trim and control was estimated as 1.5 percent of the download.

Takeoff gross weights for the twin-engine aircraft are shown in Figure 70. As was true for the single-engine helicopters, the optimum powerplant for these aircraft, from the standpoint of aircraft gross weight, would be the regenerative engine with approximately a 0.65 effectiveness regenerator.

Figures 71, 72, 73, and 74 present three-view general arrangement layout drawings of the four helicopters.

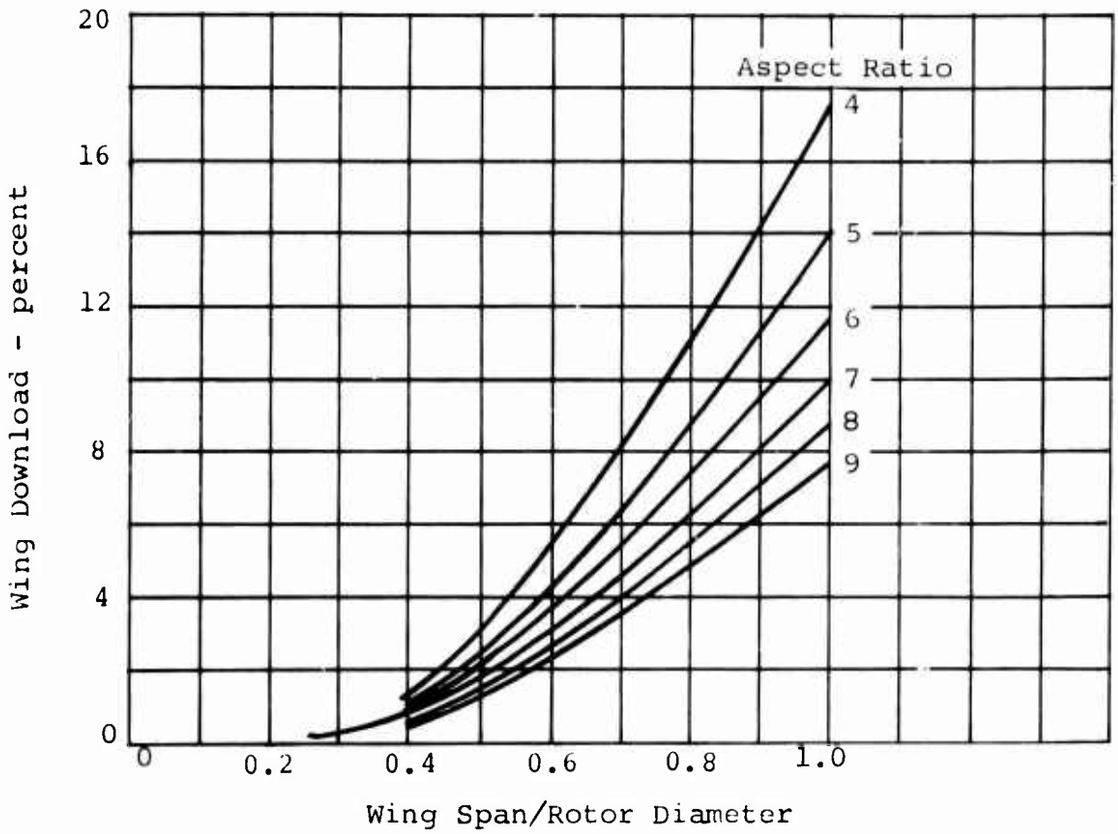


Figure 69. Single-Rotor Helicopter Wing Download.

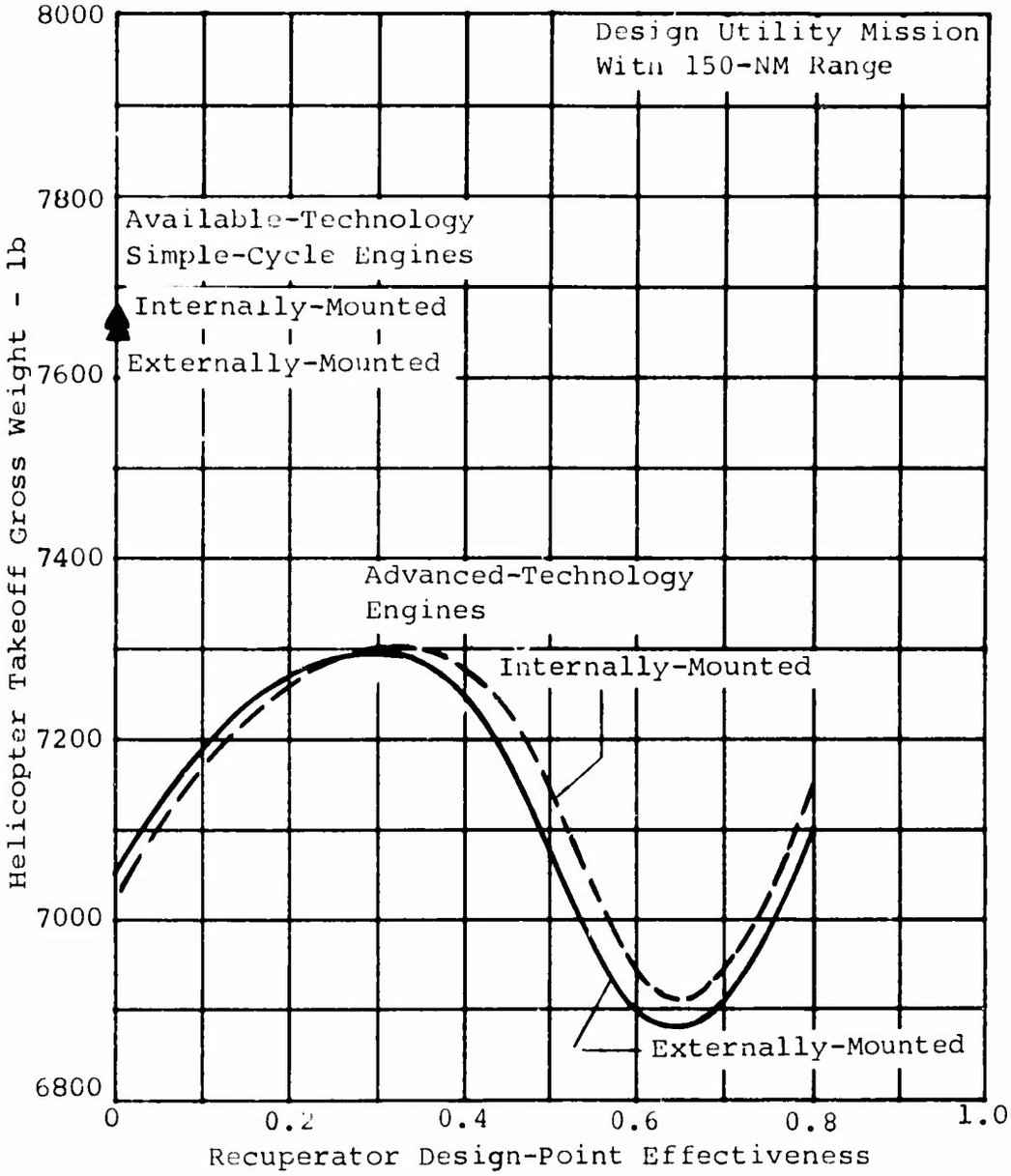


Figure 70. Takeoff Gross Weights for Twin-Engine Aircraft With Regenerative and Nonregenerative Engines.

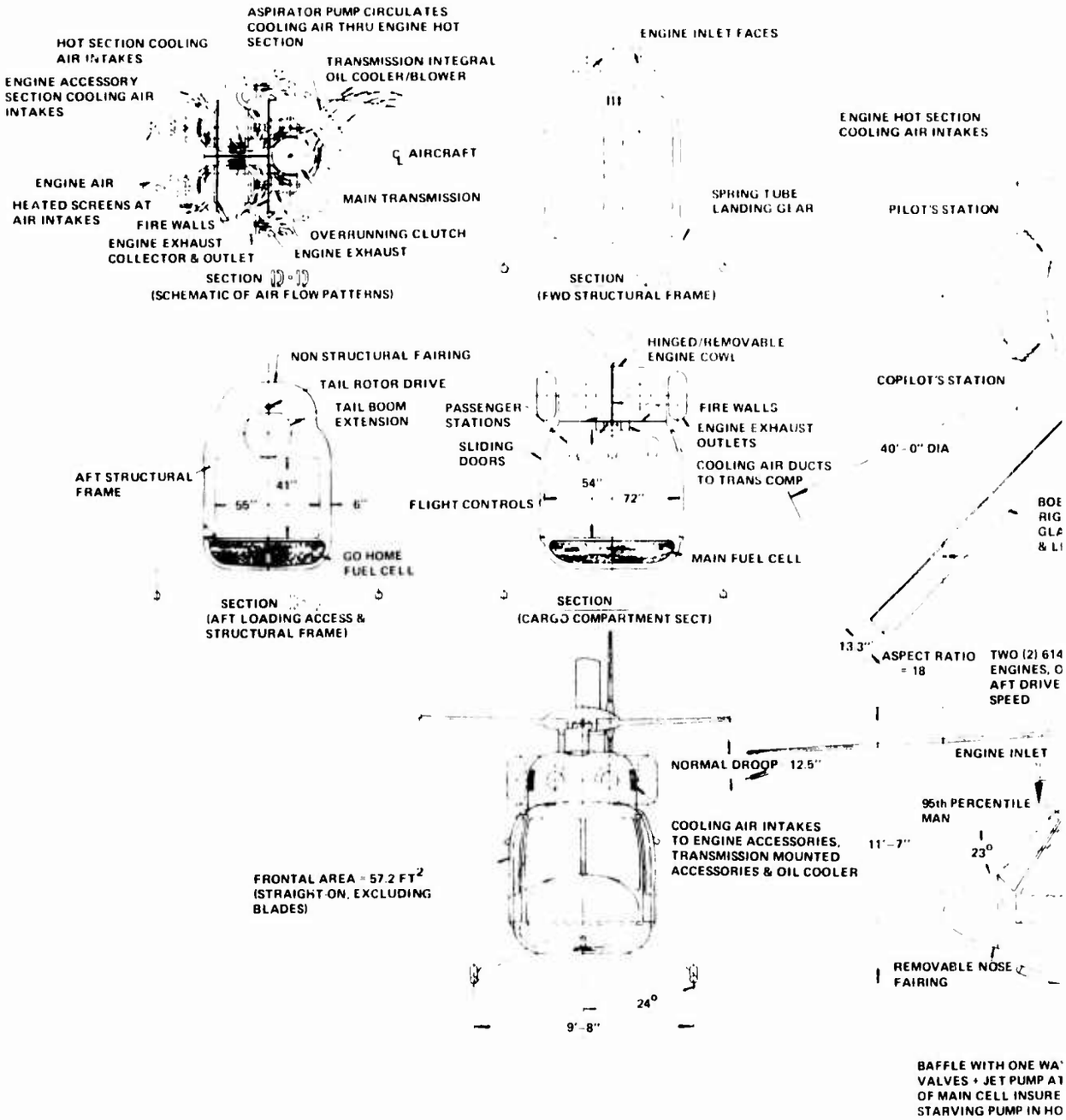
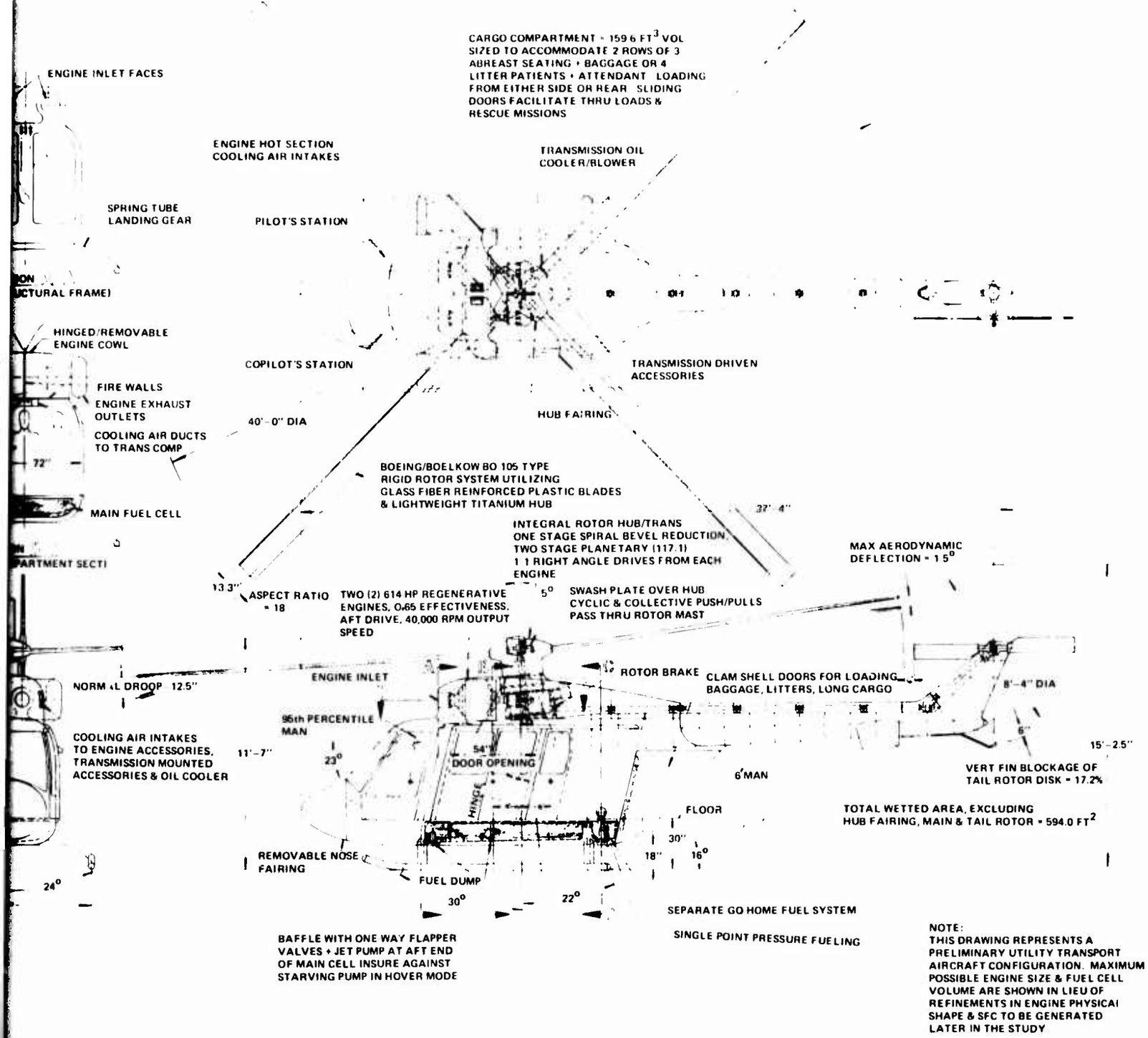


Figure 71. General Arrangement, Utility Helicopter With Advanced-Technology Regenerative Engines - Twin-Engine, Internally Mounted.

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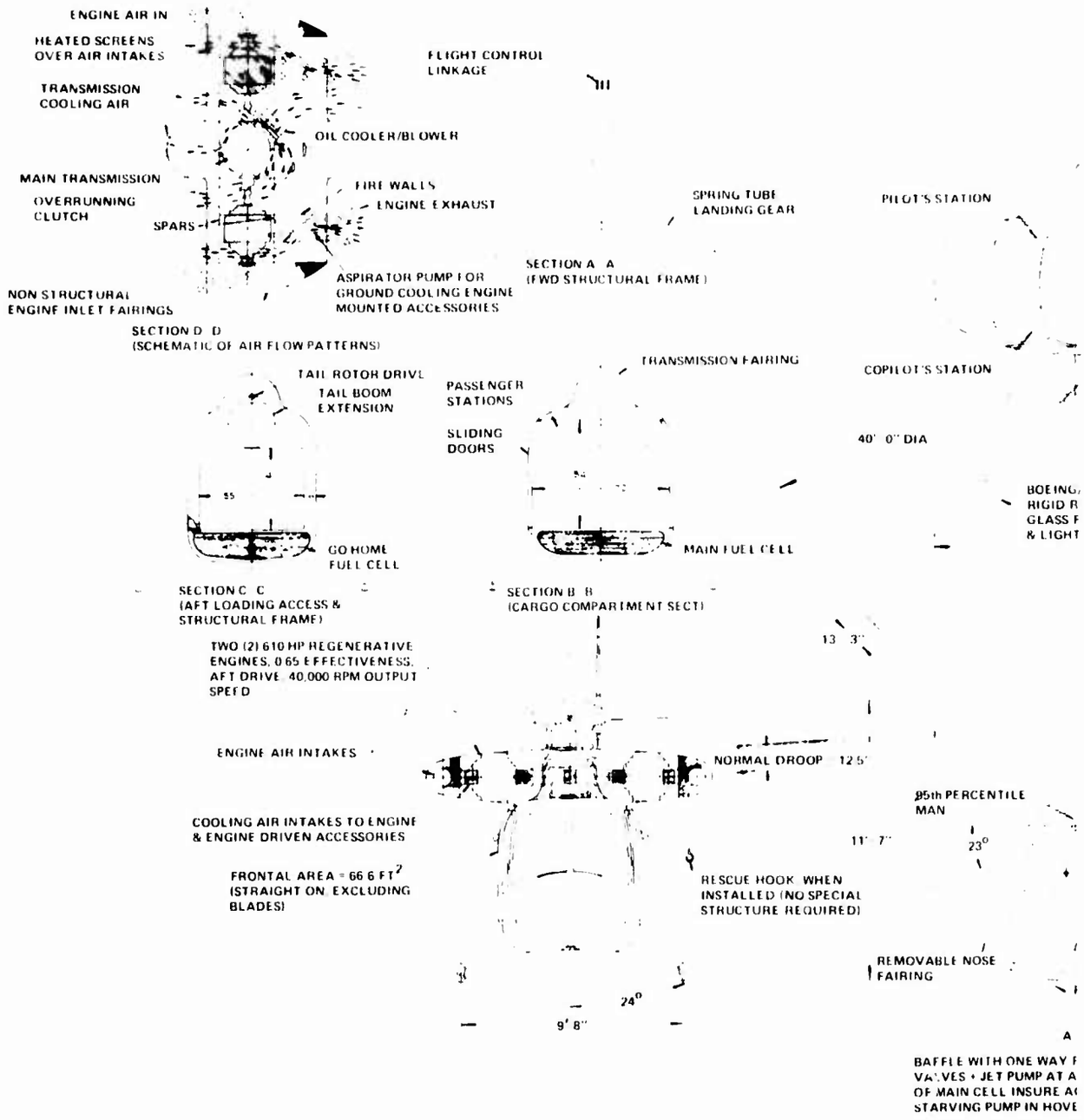
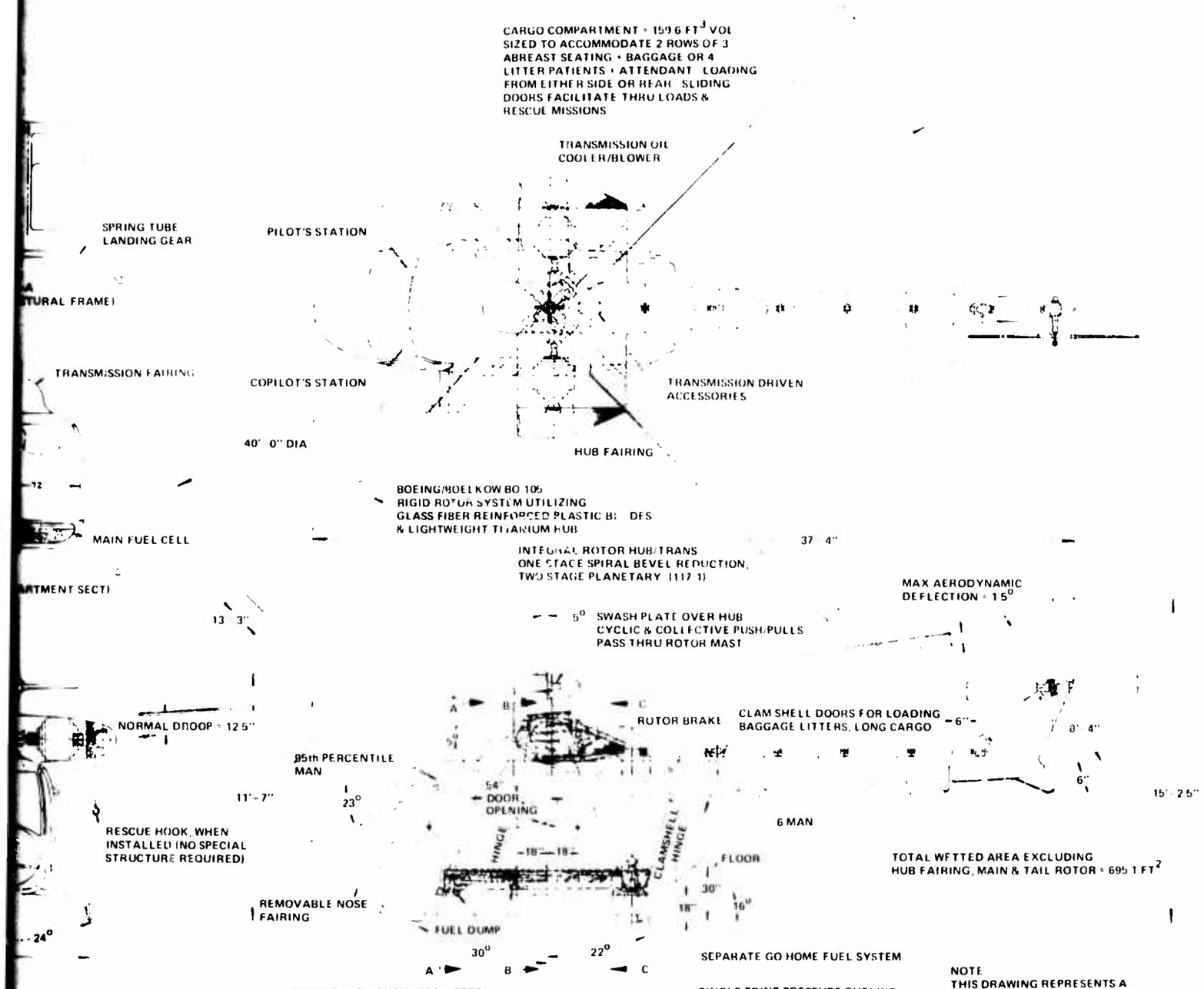


Figure 72. General Arrangement, Utility Helicopter With Advanced-Technology Regenerative Engines - Twin-Engine, Externally Mounted.



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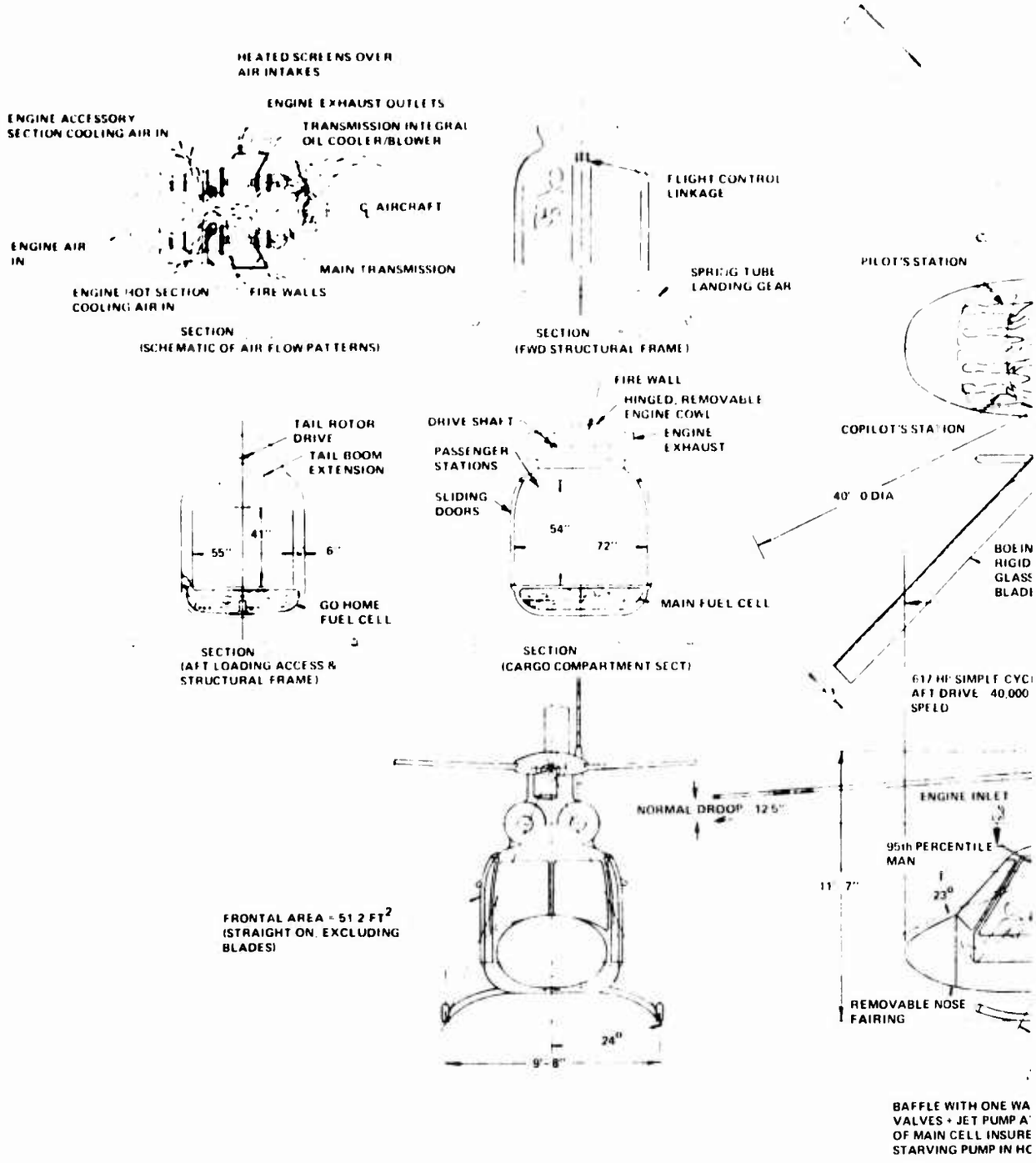
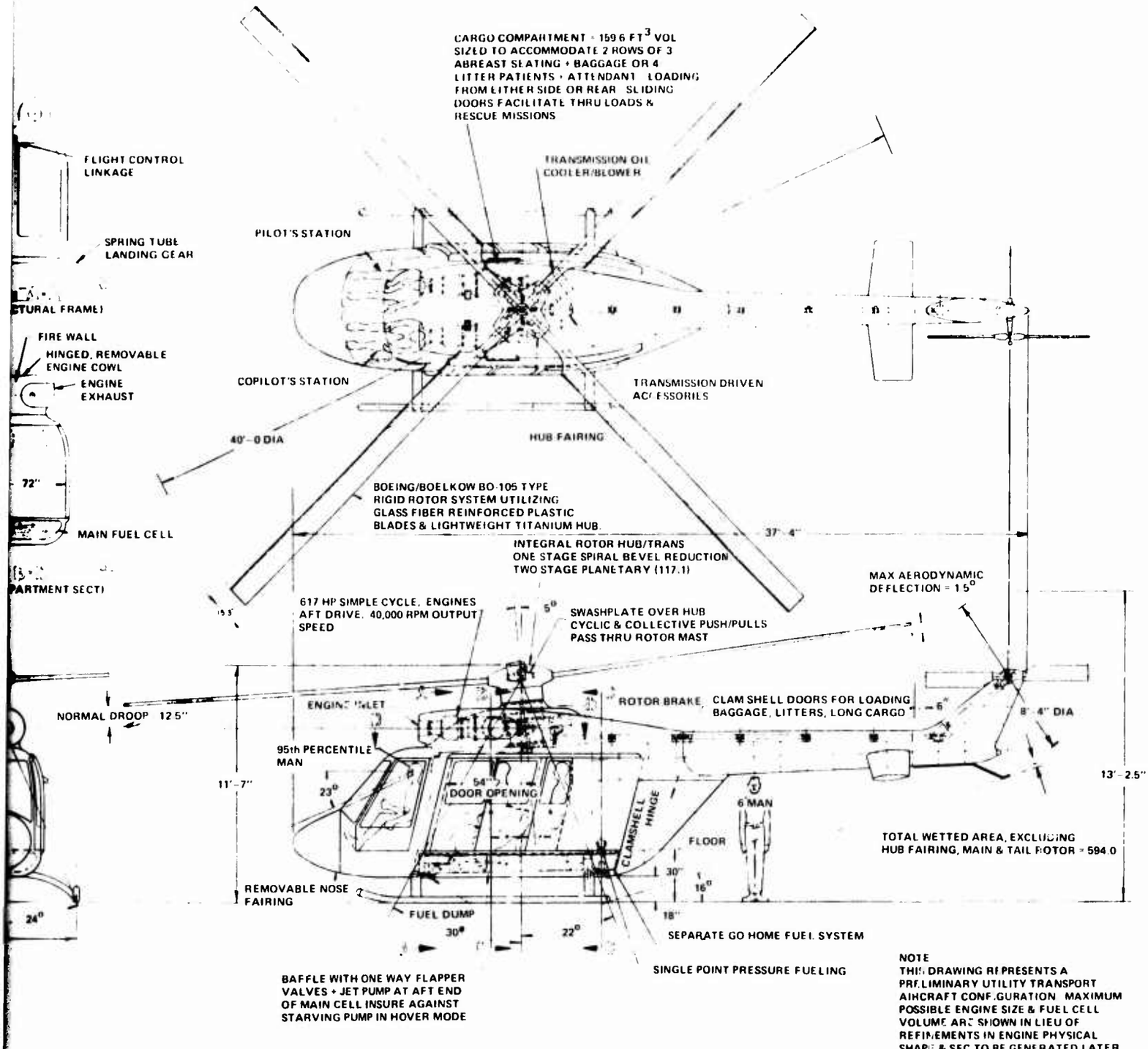


Figure 73. General Arrangement, Utility Helicopter With Simple-Cycle Engines - Twin-Engine, Internally Mounted.



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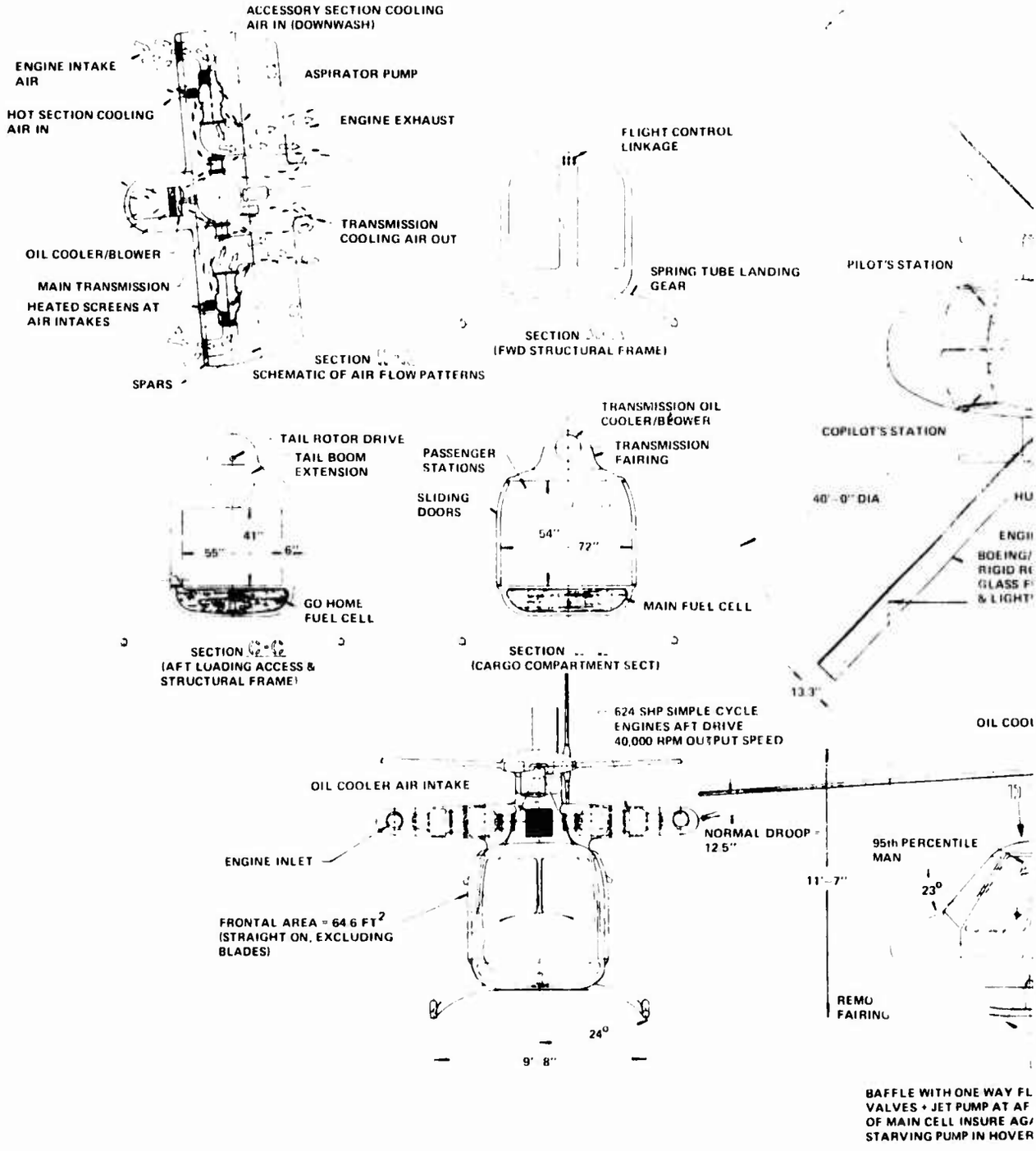
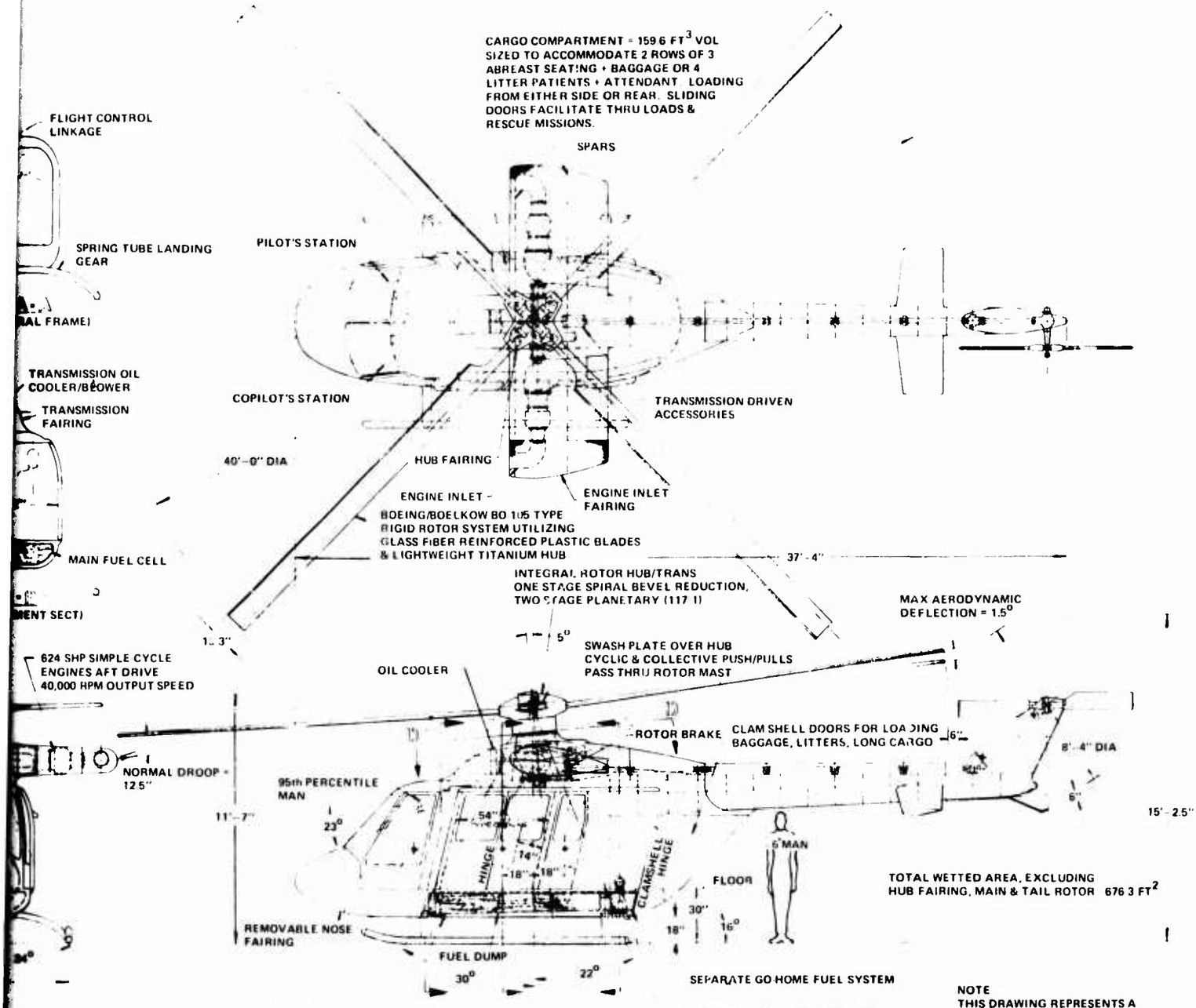


Figure 74. General Arrangement, Utility Helicopter With Simple-Cycle Engines - Twin-Engine, Externally Mounted.



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APPENDIX II

VARIABLE TURBINE GEOMETRY CONFIGURATIONS

Because the maximum benefit in engine SFC which can be realized from regeneration is related to the difference between compressor and turbine discharge temperatures, it is desirable to maintain high turbine-inlet temperature throughout a significant part of the operating range of the engine. At part-power operating points, the reduced engine airflow would contribute to improved recuperator effectiveness as well as reduced core pressure losses and, coupled with the higher turbine-inlet temperature, would produce a substantial improvement in part-power SFC. To permit part-power operation at constant turbine-inlet temperature, however, requires variable flow characteristics for the turbine, which could be accomplished with variable stator vanes. Using only variable power turbine stator vanes, the engine operating line for constant turbine-inlet temperature would be limited, to some degree, and would be highly dependent on compressor map matching characteristics, including degradation in efficiency and surge margin. For this reason, the studies of variable turbine geometry considered two regimes of operation at two different discrete turbine temperatures. The first regime considered use of variable turbine nozzles to maintain constant turbine-inlet temperature through that part-power region which could be accommodated without encountering compressor surge. For the remaining part-power regime, the turbine-inlet temperature was reduced by a fixed increment and then remained constant to very low powers.

The impact on performance of using variable geometry in both gas-generator and power-turbine nozzles was included in the study.

DESIGN-POINT PERFORMANCE

Design-point performance parameters for the variable-geometry regenerative engines were based on the component data used for the fixed-geometry engines. However, the engine operating line on the compressor performance map for the variable-geometry engine was necessarily different than the one for the fixed-geometry engine. Using the same compressor map developed previously for the regenerative engines (Figure 75), various engine operating lines have been drawn. The operating line numbered 1 corresponds to the fixed-geometry engine, and the

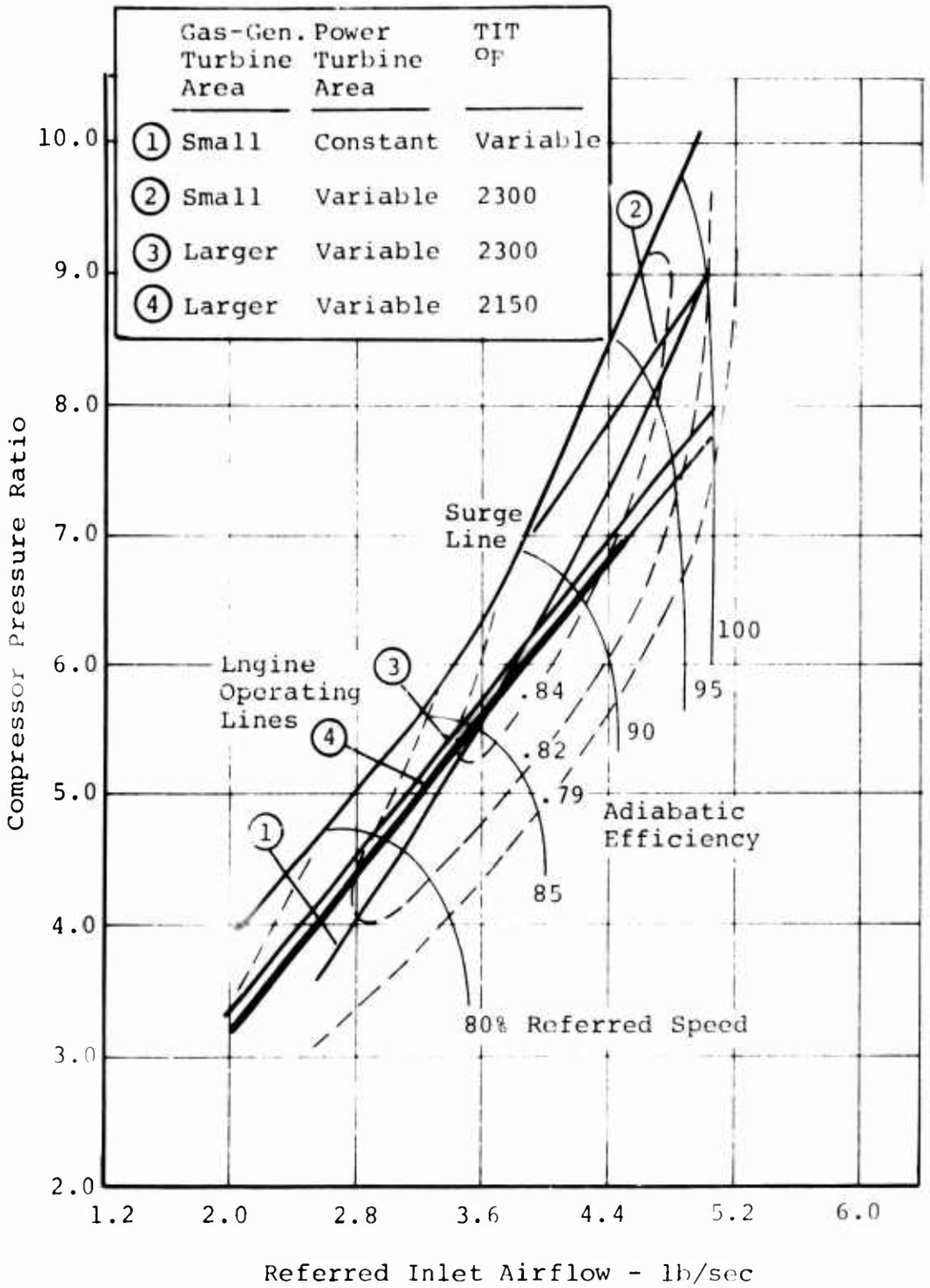


Figure 75. Compressor Performance Map for Regenerative Engines With Different Turbine Configurations.

line numbered 2 corresponds to the engine with a variable power turbine operating at 2300°F turbine-inlet temperature and passing through the 9:1 compressor pressure ratio design point. The latter operating line would intersect the surge line at part power. The remaining operating lines (numbered 3 and 4) were defined for a larger constant gas-generator turbine stator area, variable power turbine stator areas, and 2150°F and 2300°F turbine-inlet temperatures. To maintain adequate surge margin at part-power operating points (essentially the same margin as the fixed-geometry engine), it was necessary to reduce the design pressure ratio to 8:1 at the turbine-inlet temperature of 2300°F (line 3).

Design-point component performance parameters for variable- and fixed-geometry regenerative engines are compared in Table XXVI. With the exception of the compressor, design-point performance of the combustor, turbines, diffuser, and recuperator was kept identical with previous analyses. Design-point performance of the variable-geometry engine was shown to be not as good as the fixed-geometry configuration, mainly because the lower pressure ratio contributed to a slight increase in SFC and caused the specific power to be further from the optimum value.

VARIABLE POWER TURBINE STATOR (PART POWER)

Part-power performance was determined for variable power-turbine nozzle settings (changes in flow area) with fixed gas-generator turbine size (using the larger gas-generator turbine nozzle area) for two discrete values of turbine-inlet temperature throughout the engine operating range. Two turbine-temperature regimes were selected: 2300°F for use above the Normal Rated Power condition and 2150°F for use below NRP. The difference of 150°F between the two levels was consistent with the previous fixed-geometry study shown in the main text of the report.

The compressor characteristic is shown in Figure 76, with the operating lines for the two selected turbine-temperature regimes. Below NRP, the engine operating line shifts to the lower turbine-inlet temperature line. Turbine characteristics were obtained from Reference 14. The reference provided typical turbine efficiencies as a function of pressure ratio and speed, and also presented the effect of variation in power turbine nozzle area. These data were used to generate power turbine efficiency characteristics used in this study

TABLE XXVI. DESIGN-POINT PERFORMANCE PARAMETERS FOR FIXED-GEOMETRY AND VARIABLE-GEOMETRY REGENERATIVE ENGINES

	Variable Geometry	Fixed Geometry
Recuperator Effectiveness*	.65	.65
Compressor		
Inlet Airflow, lb/sec	5.0	5.0
Pressure Ratio	8.0	9.0
Adiabatic Efficiency*	.81	.82
Exit Temperature, °F	570.	601.
Cooling-Air Bleed/Inlet Airflow*	.035	.035
Leakage/Inlet Airflow	.015	.015
Recuperator - Air-Side		
Inlet Flow, lb/sec	4.75	4.75
Pressure Loss*	.024	.024
Exit Temperature, °F	1100.	1081.
Combustor		
Efficiency*	.99	.99
Fuel/Compressor Inlet Airflow	.0192	.0195
Pressure Loss*	.03	.03
Gas-Generator Turbine		
Inlet Temperature, °F	2300.	2300.
Inlet Flow, lb/sec	4.846	4.848
Mechanical Efficiency*	.975	.975
Exit Temperature, °F	1856.	1828.
Adiabatic Efficiency*	.875	.875
Pressure Ratio	2.40	2.56
Interstage Turbine Diffuser		
Pressure Loss*	.03	.03
Temperature, °F (Cooling-Air Mixed)	1832.	1805.
Power Turbine		
Inlet Temperature, °F	1832.	1805.
Inlet Flow, lb/sec	4.996	4.998
Exit Temperature, °F	1372.	1329.
Adiabatic Efficiency*	.91	.91
Pressure Ratio	2.83	2.99
Exhaust Diffuser		
Pressure Ratio	1.04	1.04
Recuperator - Gas-Side		
Inlet Flow, lb/sec	4.996	4.998
Pressure Loss*	.036	.036
Exit Temperature, °F	900.	900.
Specific Power, hp/lb/sec	186.3	192.7
Shaft Power, hp	931.	964.
SFC, lb/hr/hp	.371	.364

*Efficiencies, effectiveness, pressure losses, bleed and leakage flows expressed as decimal fractions.

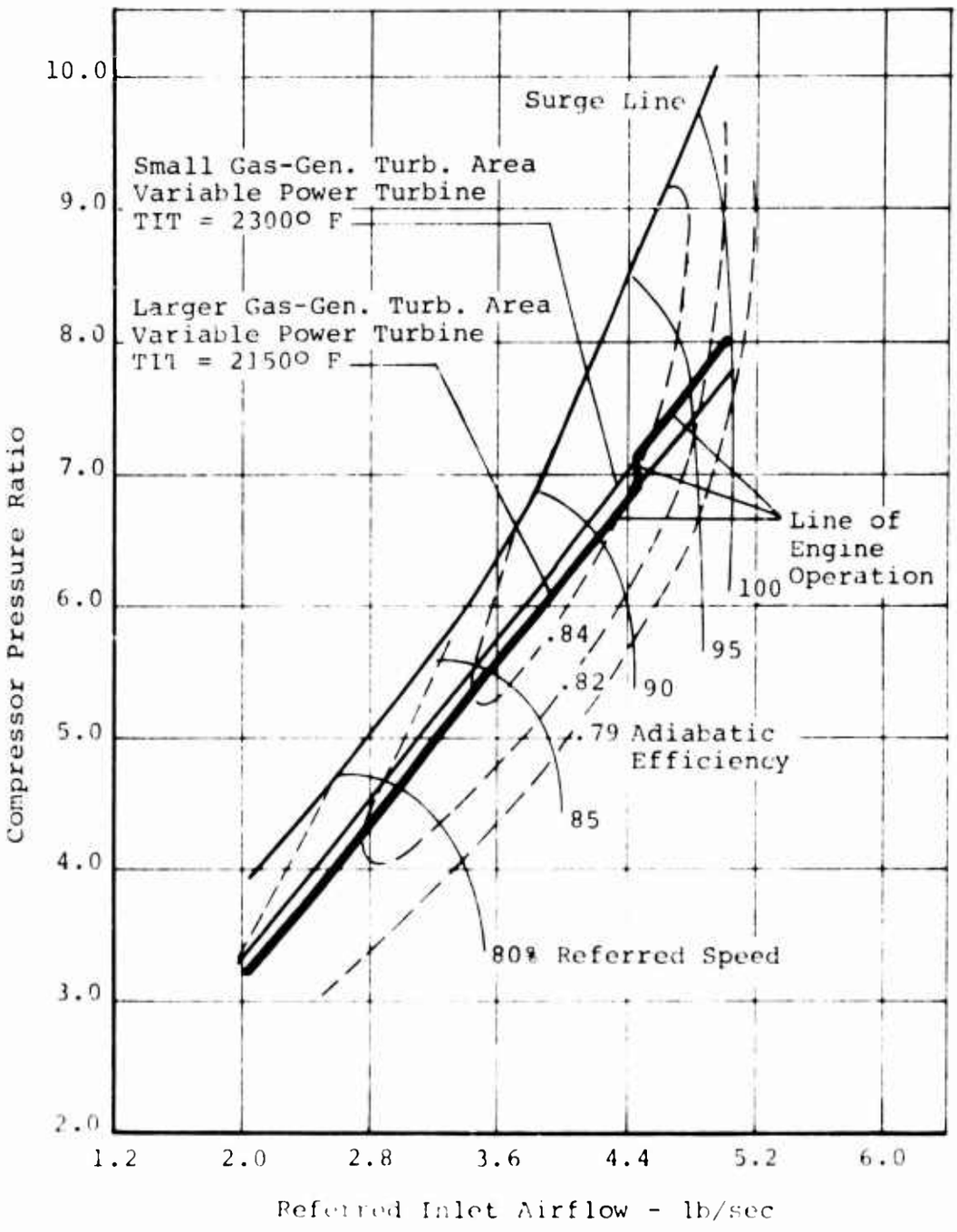


Figure 76. Compressor Performance Map for Regenerative Engine With Variable Power Turbine Stator Vanes, Constant Turbine-Inlet Temperature Operation.

(Figure 77). Off-design performance of the recuperator in terms of effectiveness and pressure loss was assumed to be the same as pictured earlier (Figures 36 and 37 in the main text of this report). Bleed and leakage flows were considered to be constant percentages of inlet airflow.

The variation in compressor airflow and the required variation in power-turbine area as a function of engine power are presented in Figure 78, for sea-level, standard-day operation.

Figure 79 shows specific fuel consumption plotted as a function of power for the regenerative engine with a variable power-turbine nozzle. Curves are plotted for the 2150°F and 2300°F turbine inlet temperature regimes, and the line of engine operation is indicated - shifting from the 2300°F curve to the 2150°F curve below NRP. These data were plotted for constant output-shaft speed, typical of the helicopter installation, and for sea level, standard day, ambient conditions.

In the specific fuel consumption plot of Figure 80, the line of engine operation for the regenerative engine with variable power-turbine geometry is compared with similar curves for a fixed-geometry regenerative engine having the same recuperator design-point effectiveness and for an advanced-technology nonregenerative engine. The variable-geometry regenerative engine SFC curve is for constant output-shaft speed. The two curves for the fixed-geometry regenerative engines also are for constant output-shaft speed. One of these curves corresponds to an engine configuration whose design output-shaft speed is the optimum power-turbine speed at Military Rated Power. The other engine's design speed is significantly lower and is equivalent to the optimum power-turbine speed at 70 percent MRP. The SFC curve for the latter engine is better at part power, although the performance at MRP is worse. For comparison, the SFC curve for the advanced-technology non-regenerative engine is included - its design output-shaft speed is also optimum at a part-power condition. The regenerative engine with variable power-turbine nozzle area offers significant performance improvement at part-power operating conditions, even when compared with a fixed-geometry regenerative engine optimized for part-power.

A curve of part-power airflow for a fixed-geometry regenerative engine was included in Figure 78. Comparing the airflow for the fixed- and variable-geometry engines, the lower airflow for the variable-geometry engine contributes to improved

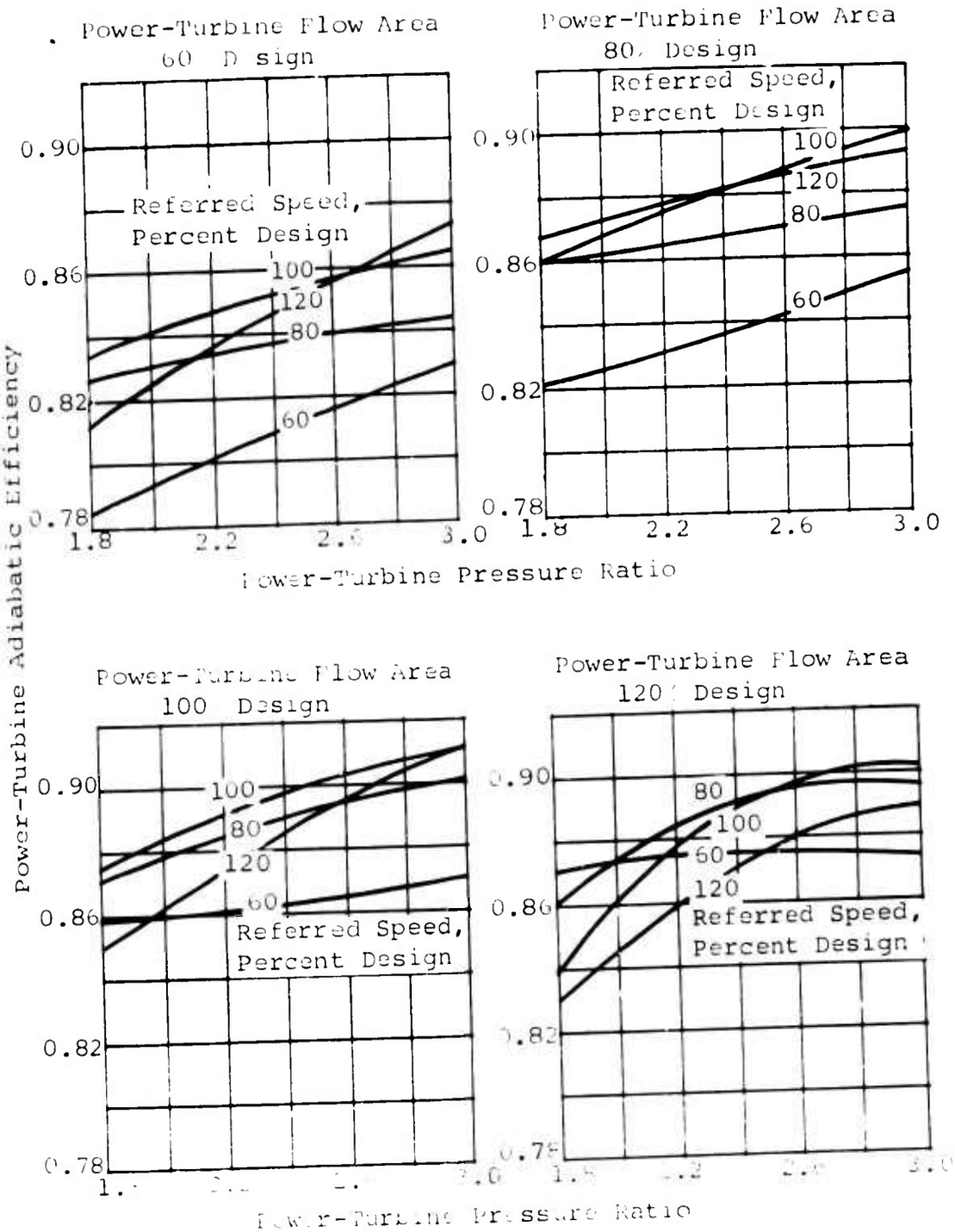


Figure 77. Performance Characteristics of Power Turbine With Variable Stator Vanes (Reference 14).

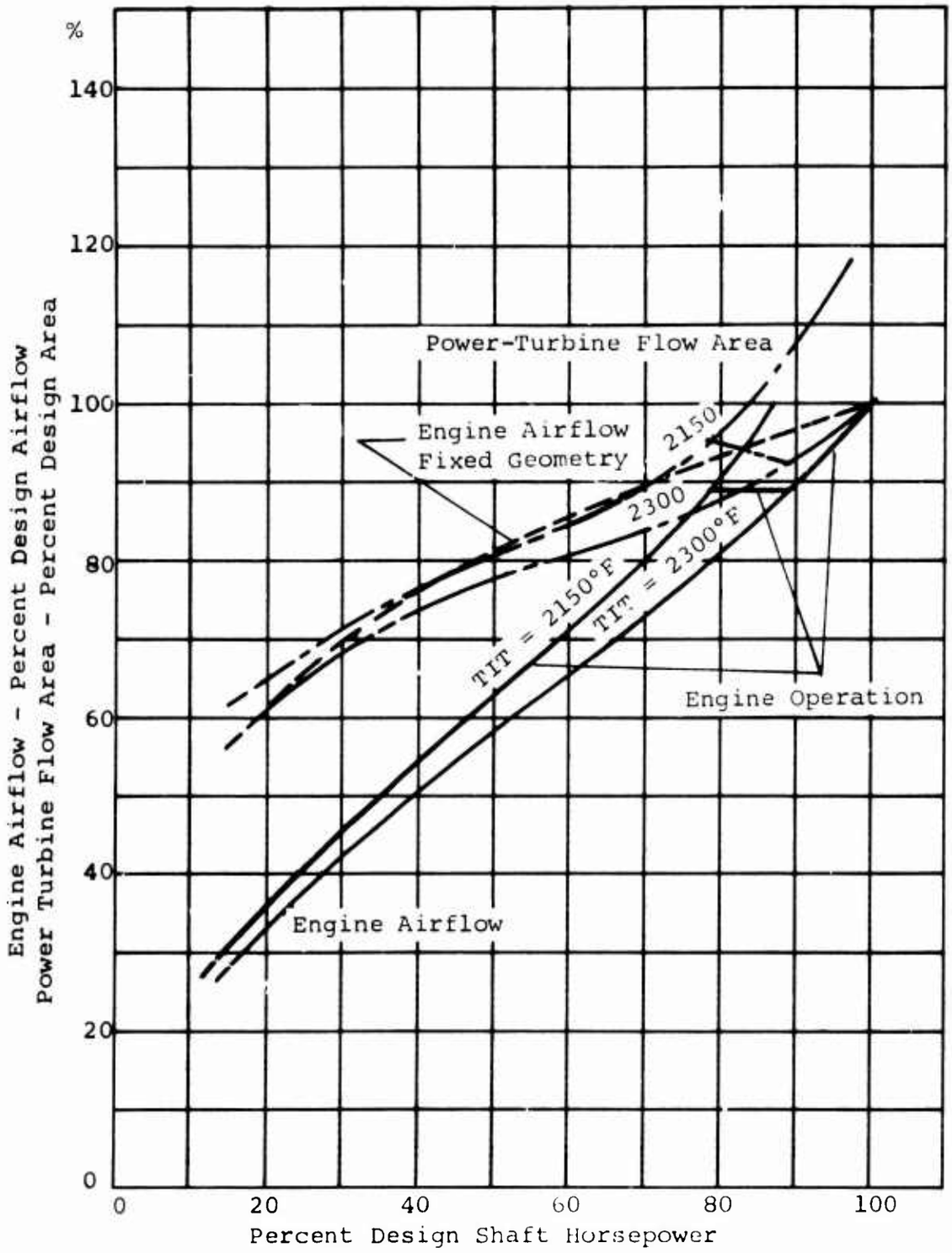


Figure 78. Variation in Engine Airflow, Power-Turbine Flow Area for Part-Power Operation of Regenerative Engine With Variable Power Turbine Stator Vanes.

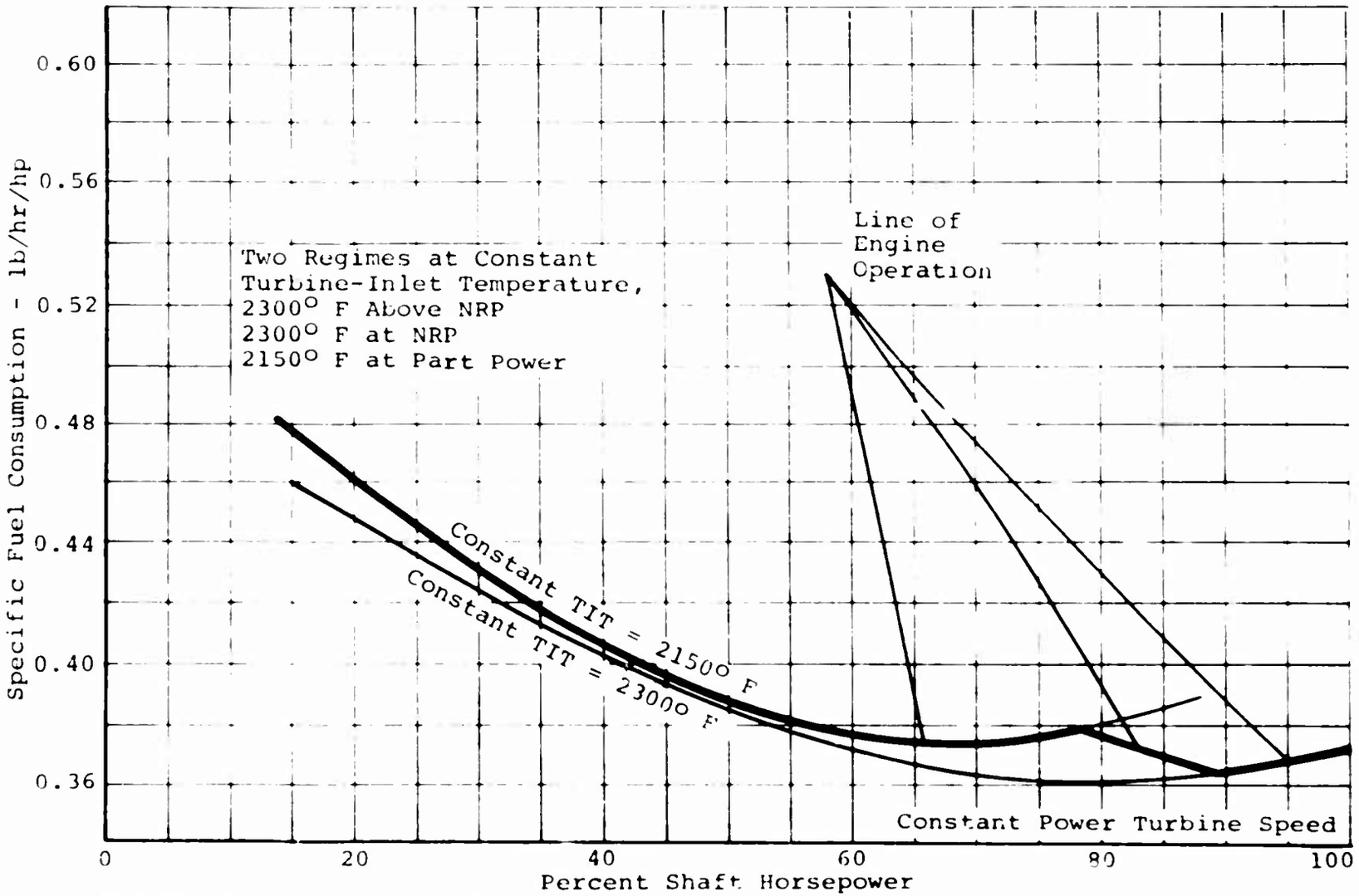


Figure 79. Specific Fuel Consumption of a Regenerative Engine With Variable Power-Turbine Flow Area.

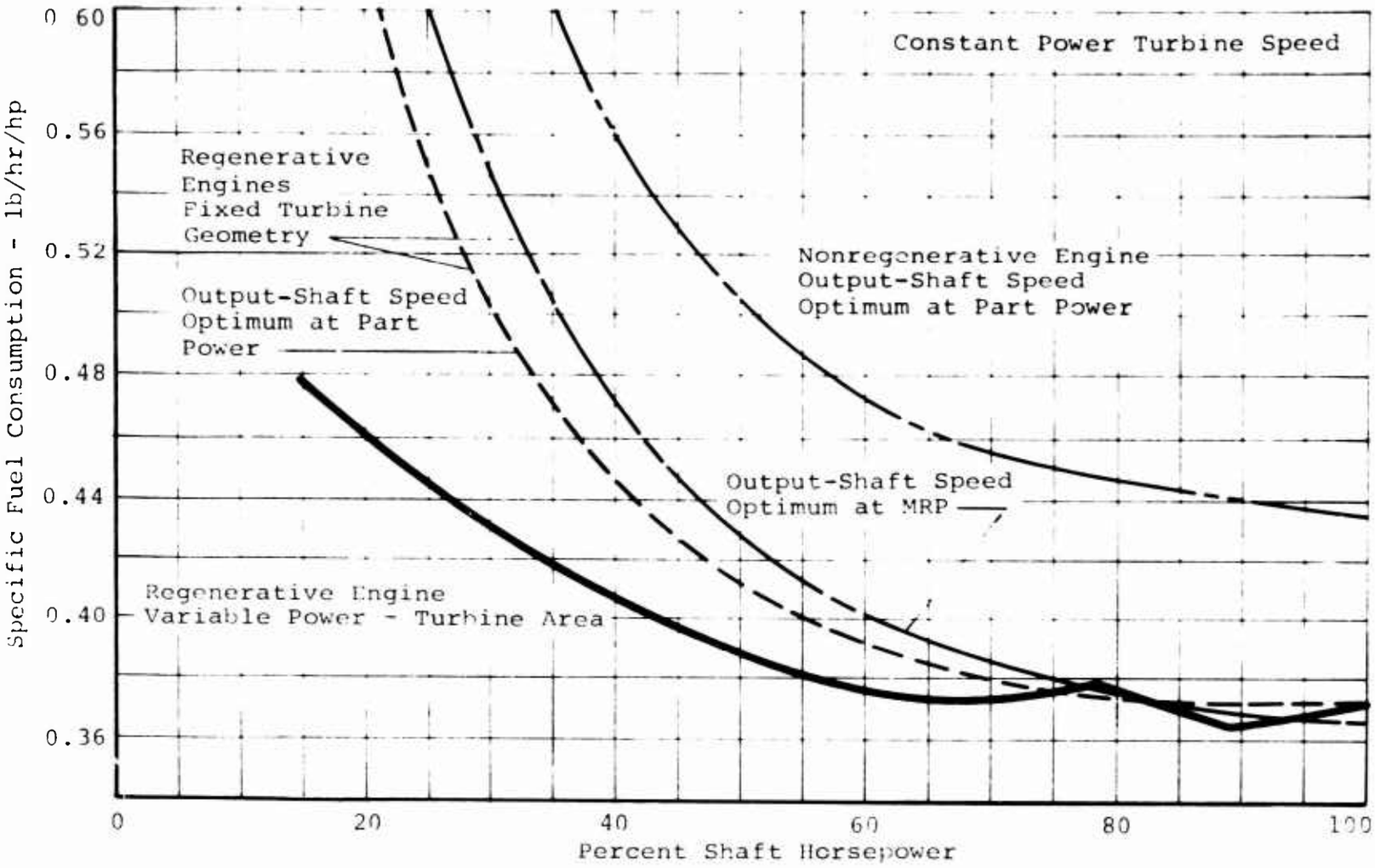


Figure 80. Specific Fuel Consumption for Regenerative Engines, With Variable Power-Turbine Flow Area and With Fixed Turbine Geometry, and for a Simple-Cycle Engine.

recuperator performance and lower SFC.

Figure 81 pictures another advantage of the variable-geometry regenerative engine. Military and Normal Rated Power are shown as a function of ambient temperature for the variable-geometry engine, and a comparable MRP point was plotted for the fixed-geometry engine at 95°F ambient temperature. The fixed-geometry engine would be turbine-inlet-temperature limited, and gas-generator speed must decrease with increasing ambient temperature. If the variable-geometry engine were gas-generator speed limited above 59°F, this engine would have a power advantage on a hot day. Further hot-day power augmentation would be possible if gas-generator speed were allowed to increase, with the power-turbine area increasing to maintain constant turbine-inlet temperature. However, a particular limitation of the compressor characteristic used for these regenerative engines was the closeness of the speed lines near the design point, typical of the choking condition encountered in highly-loaded compressors. The result is a flatter slope of the airflow-speed curve near the design point than would be the case with a more lightly loaded compressor. This difference, in turn, limited the ability of variable turbine geometry to augment hot-day power. A variable-geometry engine with a more favorable compressor characteristic would provide greater improvement (compared to fixed geometry) on a 95°F day if the gas-generator speed were not limited.

Figure 80 shows that the variable-geometry regenerative engine offers fuel savings for cruise at high power settings, but the gain is almost negligible. Only those missions which required long loiter times at minimum-power speed would result in substantial fuel savings compared to the fixed-geometry engine, enough to offset the increased weight and complexity associated with the variable turbine nozzle vane concept.

VARIABLE GAS-GENERATOR AND POWER TURBINES

With variable nozzle vanes in both the gas-generator turbine and power turbine, it could be possible to operate in a region of good compressor efficiency away from the surge line for a substantial portion of the operating range of the engine. This concept would introduce an added degree of complexity to the control requirements; however, the impact on performance of using variable geometry in both turbines was investigated. In Figure 82, the same regenerative engine compressor characteristic is shown, with operating lines defined by

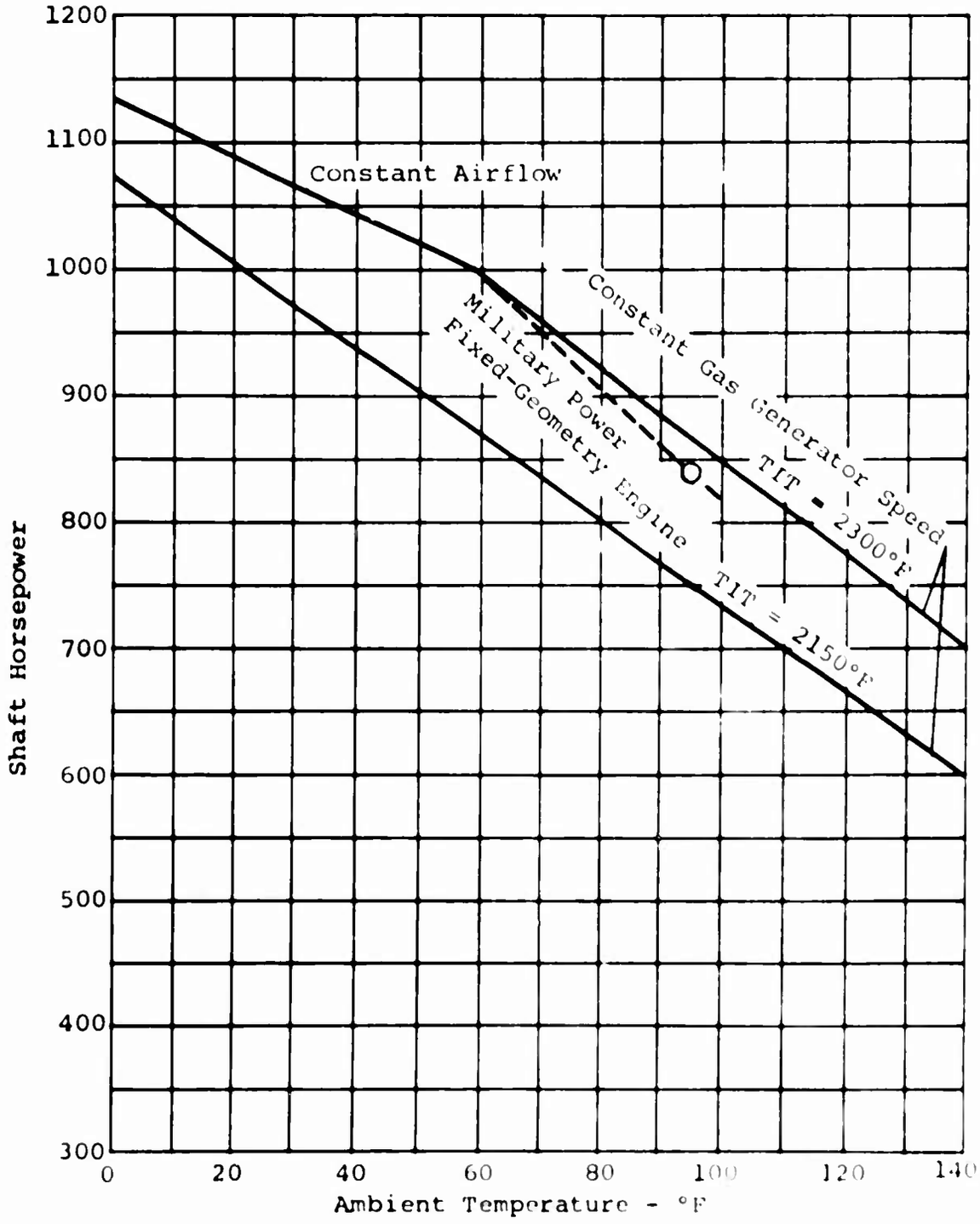


Figure 81. Military and Normal Power as a Function of Ambient Temperature for Regenerative Engine With Variable Power-Turbine Area.

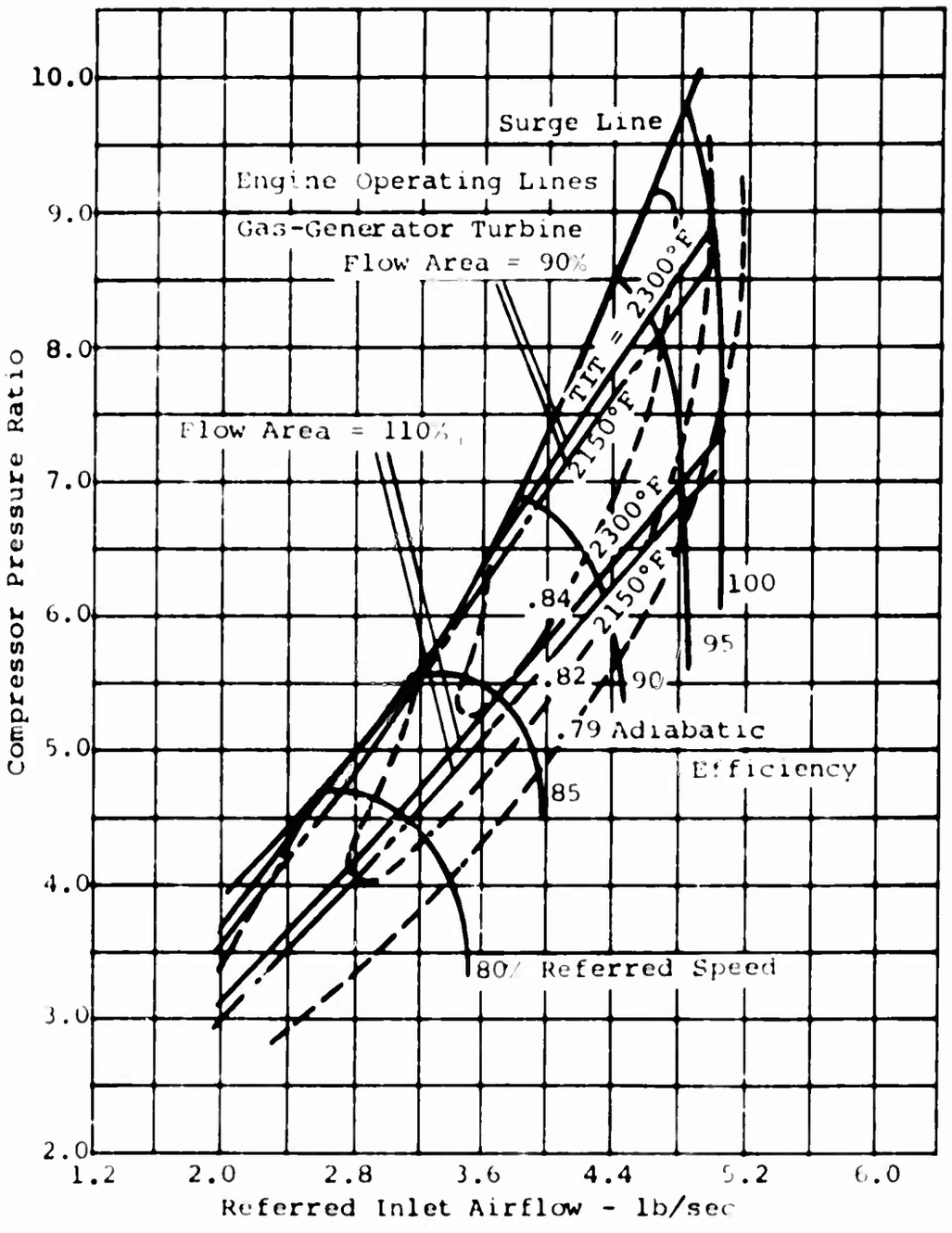


Figure 82. Compressor Performance Map for Regenerative Engine, and Operating Lines for Variable Gas-Generator and Power Turbines, Constant Turbine-Inlet Temperature.

different values of gas-generator turbine area again for the two turbine-temperature regimes previously selected. The power turbine area was varied to establish points along operating lines for selected gas-generator nozzle settings. Performance characteristics for both turbines were determined from the turbine-efficiency trend data in Reference 14.

Engine performance corresponding to the various operating lines on the compressor map is presented in terms of SFC as a function of power in Figure 83. Curves are plotted for different gas-generator turbine areas, and the power turbine area varies continually along each curve. These curves illustrate that performance improvements with added variable gas-generator turbine geometry are negligible and would not warrant the additional complexity of this concept.

Although it was not possible to obtain quantified data on the impact of variable power-turbine nozzle vane concepts on engine weight and cost, the previous figures showed negligible fuel savings at high power settings, typical of the helicopter flying a utility mission. Generally, any fuel saving was not large enough to warrant the added weight and complexity of variable-turbine designs. Only missions requiring long loiter times at minimum power speed offer potential for the variable-geometry regenerative engine.

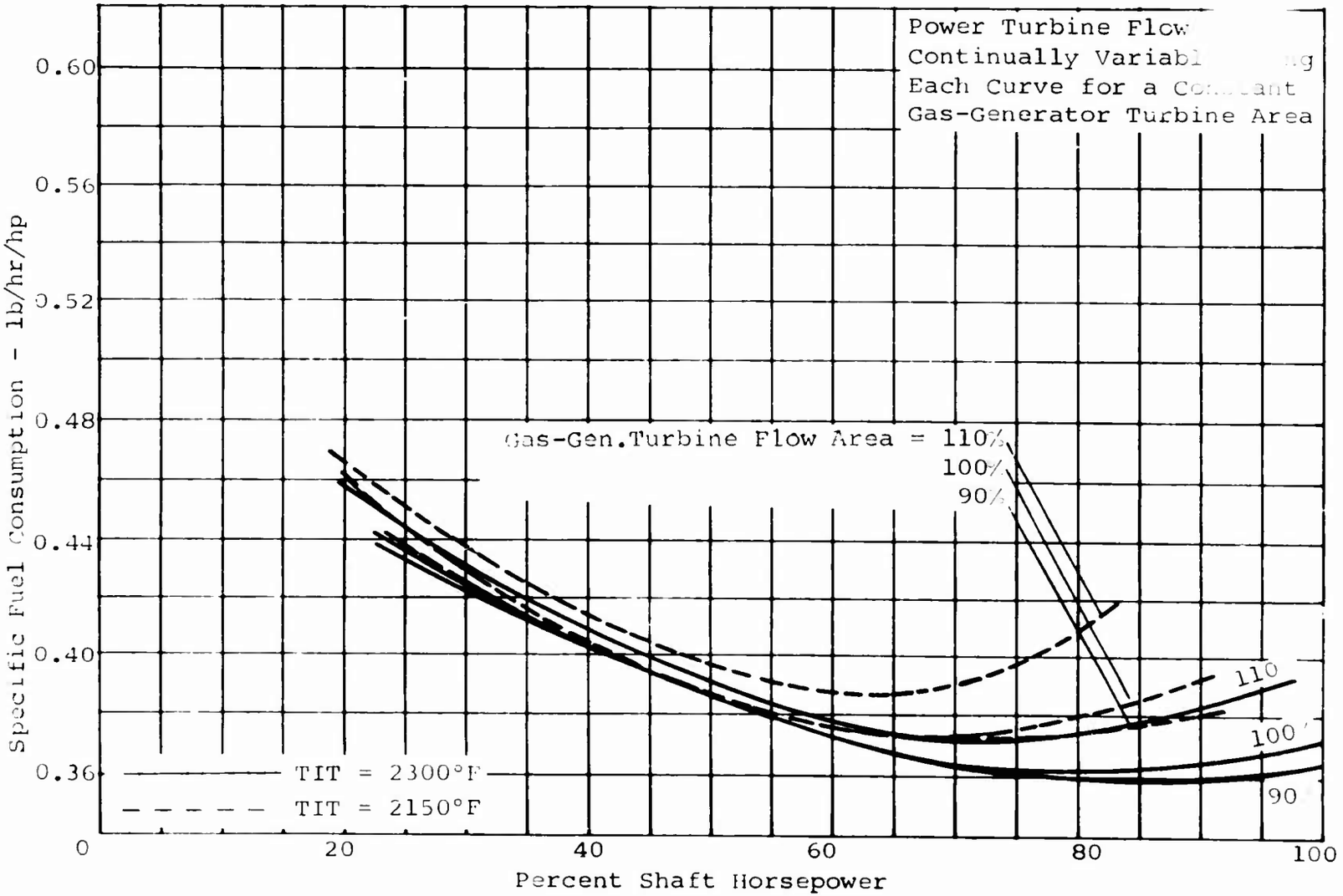


Figure 83. Specific Fuel Consumption of Regenerative Engines With Variable Gas Generator Turbine and Power-Turbine Flow Areas.

APPENDIX III

SUMMARY WEIGHT STATEMENTS

MIL-STD-451, Part I
NAME J. F. Biglin, Jr.
DATE

PAGE
MODEL
REPORT

SUMMARY WEIGHT STATEMENT
ROTORCRAFT ONLY
ESTIMATED - ~~CALCULATED~~ - ~~ACTUAL~~
(Cross out those not applicable)

Advanced-Technology Regenerative Engines
Advanced-Technology Simple-Cycle Engine
Available-Technology Simple-Cycle Engine

CONTRACT DAAJ02-70-C-0061
ROTORCRAFT, GOVERNMENT NUMBER
ROTORCRAFT, CONTRACTOR NUMBER
MANUFACTURED BY The Boeing Company, Vertol Division

		Main	Auxiliary
Engine	Manufactured by	AiResearch	
	Model		
	Number		
Propeller	Manufactured by		
	Model		
	Number		

	Advanced-Technology Regenerative Engines						Simple-Cycle Engines			
	40% Effectiveness		65% Effectiveness		80% Effectiveness		Advanced-Technology		Available-Technology	
ROTOR GROUP		630		624		628		628		646
BLADE ASSEMBLY	320		312		314		314		323	
HUB	310		312		314		314		323	
HINGE AND BLADE RETENTION										
FLAPPING										
LEAD LAG										
PITCH										
FOLDING										
WING GROUP										
WING PANELS-BASIC STRUCTURE										
CENTER SECTION-BASIC STRUCTURE										
INTERMEDIATE PANEL-BASIC STRUCTURE										
OUTER PANEL-BASIC STRUCTURE-INCL TIPS LBS										
SECONDARY STRUCT-INCL FOLD MECH LBS										
AILERONS - INCL BALANCE WTS LBS										
FLAPS										
-TRAILING EDGE										
-LEADING EDGE										
SLATS										
SPOILERS										
TAIL GROUP		102		100		102		97		106
TAIL ROTOR	31		30		31		30		33	
- BLADES										
- HUB										
STABILIZER - BASIC STRUCTURE	31		31		31		31		32	
FINS - BASIC STRUCTURE - INCL DORSAL LBS	40		39		40		36		41	
SECONDARY STRUCTURE - STABILIZER AND FINS										
ELEVATOR - INCL BALANCE WEIGHT LBS										
RUDDER - INCL BALANCE WEIGHT LBS										
BODY GROUP		1129		1117		1123		1135		1158
FUSELAGE OR HULL - BASIC STRUCTURE										
BOOMS - BASIC STRUCTURE										
SECONDARY STRUCTURE - FUSELAGE OR HULL										
- CABIN										
- DOORS, PANELS & MISC										
ALIGHTING GEAR - LAND TYPE		146		144		145		146		155
LOCATION LANDING GEAR ASSEMBLY	STRUCT CONTROLS									
ALIGHTING GEAR GROUP - WATER TYPE										
LOCATION FLOATS	STRUTS CONTROLS									

* WHEELS, BRAKES, TIRES, TUBES AND RIM

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NAME SUMMARY WEIGHT STATEMENT MODEL
DATE WEIGHT EMPTY REPORT

	Advanced-Technology Regenerative Engines						Simple-Cycle Engines			
	40% Effectiveness		65% Effectiveness		80% Effectiveness		Advanced-Technology		Available-Technology	
FLIGHT CONTROLS GROUP		325		323		324		324		333
COCKPIT CONTROLS		20		20		20		20		20
AUTOMATIC STABILIZATION		25		25		25		25		25
SYSTEM CONTROLS - ROTOR NON ROTATING		120		119		119		119		122
ROTATING		160		159		160		160		165
- FIXED WING										
ENGINE SECTION OR NACELLE GROUP		75		77		90		71		71
INBOARD										
CENTER										
OUTBOARD										
DOORS, PANELS AND MISC										
PROPULSION GROUP		1268		1230		1309		1225		1489
X AUXILIARY X X MAIN X										
ENGINE INSTALLATION										
ENGINE		174		198		285		142		296
TIP BURNERS										
LOAD COMPRESSOR										
REDUCTION GEAR BOX, ETC										
ACCESSORY GEAR BOXES AND DRIVES										
SUPERCHARGER-FOR TURBOS										
AIR INDUCTION SYSTEM		35		35		35		30		35
EXHAUST SYSTEM		15		15		20		11		11
COOLING SYSTEM		10		10		10		5		5
LUBRICATING SYSTEM		5		5		5		5		5
TANKS	-		-		-		-		-	
BACKING BD.TANK SUP & PADDING	-		-		-		-		-	
COOLING INSTALLATION	-		-		-		-		-	
PLUMBING, ETC	5		5		5		5		5	
FUEL SYSTEM		455		405		385		461		526
TANKS - UNPROTECTED										
- PROTECTED	353		303		283		359		424	
BACKING BD.TANK SUP & PADDING	67		67		67		67		67	
PLUMBING, ETC	35		35		35		35		35	
WATER INJECTION SYSTEM										
ENGINE CONTROLS		10		10		10		10		10
STARTING SYSTEM (Incl. in Electrical Group)		-		-		-		-		-
PROPELLER INSTALLATION		-		-		-		-		-
DRIVE SYSTEM		564		552		559		561		601
GEAR BOXES	488		477		483		485		520	
LUBE SYSTEM	44		43		44		44		47	
CLUTCH AND MISC	12		12		12		12		13	
TRANSMISSION DRIVE	80		20		20		20		21	
ROTOR SHAFT										
JET DRIVE (Incl. in Gear Boxes)										
AUXILIARY POWER PLANT GROUP										

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NAME SUMMARY WEIGHT STATEMENT
DATE WEIGHT EMPTY
PAGE
MODEL
REPORT

		Advanced-Technology Regenerative Engines						Simple-Cycle Engines			
		40% Effectiveness		65% Effectiveness		80% Effectiveness		Advanced-Technology		Available Technology	
INSTRUMENT AND NAVIGATIONAL EQUIPMENT GROUP			23		23		23		23		23
INSTRUMENTS		23									
NAVIGATIONAL EQUIPMENT											
HYDRAULIC AND PNEUMATIC GROUP											
HYDRAULIC											
PNEUMATIC											
ELECTRICAL GROUP			125		12		125		125		125
A C SYSTEM											
D C SYSTEM		125									
ELECTRONICS GROUP			108		108		108		108		108
EQUIPMENT		117									
INSTALLATION		41									
ARMAMENT GROUP - INCL GUNFIRE PROTECTION	LBS		50		50		50		50		50
FURNISHINGS AND EQUIPMENT GROUP			60		60		60		60		60
ACCOMMODATIONS FOR PERSONNEL		30		30		30		30		30	
MISCELLANEOUS EQUIPMENT & INCL	LBS BALLAST &	10		10		10		10		10	
FURNISHINGS		10		10		10		10		10	
EMERGENCY EQUIPMENT		10		10		10		10		10	
AIR CONDITIONING AND ANTI-ICING EQUIPMENT			30		30		30		30		30
AIR CONDITIONING		30									
ANTI-ICING											
PHOTOGRAPHIC GROUP											
EQUIPMENT											
INSTALLATION											
AUXILIARY GEAR GROUP											
AIRCRAFT HANDLING GEAR											
LOAD HANDLING GEAR											
ATO GEAR											
VIBRATION CONTROL DEVICES			57		57		57		57		59
CONTINGENCY ALLOWANCE			33		33		33		33		35
MANUFACTURING VARIATION											
TOTAL-WEIGHT EMPTY - PAGES 2, 3 AND 4			4211		4151		4257		4162		4424

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LOAD CONDITION			Advanced-Technology Regenerative Engines									Simple-Cycle Engines					
			60% Effectiveness			65% Effectiveness			80% Effectiveness			Advanced-Technology			Available-Technology		
			UTILITY	OBSERV.	MEDEVAC	UTILITY	OBSERV.	MEDEVAC	UTILITY	OBSERV.	MEDEVAC	UTILITY	OBSERV.	MEDEVAC	UTILITY	OBSERV.	MEDEVAC
CREW - NO. 3			600	600	600	600	600	600	600	600	600	600	600	600	600	600	600
PASSENGERS - NO. 4/1/1			800	200	200	800	200	200	800	200	200	800	200	200	800	200	200
FUEL LOCATION	TYPE	GALS															
UNUSABLE		7	7	7	7	7	7	7	7	7	7	7	7	7	7	7	7
INTERNAL		469	469	392	394	394	331	363	363	306	478	478	394	576	576	482
AUX. FUEL																	
- SYSTEM				23			21			20			22			26
EXTERNAL																	
BOMB BAY																	
OIL																	
UNUSABLE		3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3
ENGINE		10	10	10	10	10	10	10	10	10	10	10	10	10	10	10	10
BAGGAGE																	
CARGO			400	-	-	400	-	-	400	-	-	400	-	-	400	-	-
LITTER PATIENTS & LITTERS(4)			-	-	1000	-	-	1000	-	-	1000	-	-	1000	-	-	1000
ARMAMENT																	
GUNS-LOCATION	TYPE	QUANTITY CAL/SIZE															
-FUSELAGE		1 5.56 MM	12	12	12	12	12	12	12	12	12	12	12	12	12	12	12
AMM		5.56 MM	100	100	100	100	100	100	100	100	100	100	100	100	100	100	100
BOMB INSTL*																	
BOMBS																	
TORPEDO INSTL*																	
TORPEDOES																	
ROCKET INSTL*																	
ROCKETS																	
EQUIPMENT-PYROTECHNICS																	
-PHOTOGRAPHIC																	
-FROGH BOATS(4)/(1)/(1)			25	7	7	25	7	7	25	7	7	25	7	7	25	7	7
-SERVICE																	
-R.A.C. COMMUNICATION EQU.			23	-	-	23	-	-	23	-	-	23	-	-	23	-	-
-MEDICAL EQUIPMENT			-	1	3	-	1	3	-	1	3	-	1	3	-	1	3
-LITTER SUPPORTS			-	-	40	-	-	40	-	-	40	-	-	40	-	-	40
-MEDICAL EQUIPMENT			-	-	100	-	-	100	-	-	100	-	-	100	-	-	100
USEFUL LOAD			2469	1409	2505	2374	1334	2442	2343	1303	2416	2458	1418	2506	2556	1516	2598
GROSS WEIGHT - PAGE 4			4211	4211	4211	4151	4151	4151	4257	4257	4257	4162	4162	4162	4494	4494	4494
GROSS WEIGHTS - PAGES 2+3			6680	5620	6716	6525	5485	6593	6600	5560	6673	6620	5580	6668	7050	6010	7092

* IF NOT SPECIFIED AS BELOW ENTRY AS FIXED, FLEXIBLE, ETC.

	Advanced-Technology Regenerative Engines									Simple-Cycle Engines					
	40% Effectiveness			65% Effectiveness			80% Effectiveness			Advanced-Technology			Available-Technology		
1 LENGTH - OVERALL (ROTORS TURNING) 49.0 FT 4 BLADES FOLDED 37.33 FT															
2 GENERAL DATA BODY	FUS	MAC	CABIN												
3 LENGTH - MAXIMUM FEET	35.83			35.83			35.83			35.83			35.83		
4 DEPTH - MAXIMUM FEET	6.67			6.67			6.67			6.67			6.67		
5 WIDTH - MAXIMUM FEET	6.50			6.50			6.50			6.50			6.50		
6 WETTED AREA TOTAL	538.5			538.5			538.5			542.5			542.5		
7 WETTED AREA GLASS															
8 WING TAIL & FLOOR DATA WING	H TAIL	V TAIL	FLOOR												
9 GROSS AREA - SQUARE FEET															
10 WEIGHT/GROSS AREA - POUNDS PER SQUARE FEET															
11 SPAN - FEET															
12 FOLDED SPAN - FEET															
13 THEORETICAL ROOT CHORD - INCHES															
14 MAXIMUM THICKNESS - INCHES															
15 CHORD AT PLANFORM BREAK INCHES															
16 MAXIMUM THICKNESS - INCHES															
17 THEORETICAL TIP CHORD - INCHES															
18 MAXIMUM THICKNESS - INCHES															
19 DORSAL AREA INCLUDED IN FUSELAGE SQ FT	TAIL		SQ FT												
20 TAIL LENGTH 25% MAC WING TO 25% MAC HORIZONTAL TAIL		FEET													
21 AREA - SQ FT PER ROTORCRAFT FLAPS AILERONS		SPOILERS													
22 SLATS WING LE		WING TE													
23 ROTOR DATA - TYPE - ~~ARTICULATED~~ GIMBALED - ~~TELESCOPING~~		RIGID													
24 X MAIN ROTOR X	TAIL ROTOR		X	X TAIL ROTOR		X	X TAIL ROTOR		X	X TAIL ROTOR		X	X TAIL ROTOR		
25 FROM CL ROTATION - INCHES ROOT TIP	ROOT		TIP	ROOT			ROOT			ROOT			ROOT		
26 CHORD - INCHES 13.2	9.07			7.97			8.02			8.04			8.37		
27 THICKNESS - INCHES															
28	MAIN-FWD		TAIL	MAIN-FWD		TAIL	MAIN-FWD		TAIL	MAIN-FWD		TAIL	MAIN-FWD		TAIL
29 BLADE RADIUS - FEET	20.0		1.89	20.0		3.045	20.0		1.87	20.0		1.88	20.0		4.015
30 NUMBER BLADES	4		2	4		2	4		2	4		2	4		2
31 BLADE AREA-TOTAL-OUTBOARD INCHES RADIUS															
32 DISC AREA - TOTAL SWEPT 1256.6 SQUARE FEET - OVERLAP SQUARE FEET															
33 TIP SPEED AT DESIGN LIMIT ROTOR-SPEED-POWER-FT/SEC *****	766	770		766	770		766	770		766	770		766	770	
34 DESIGN FACTOR USED BY CONTRACTOR.	1.07	1.07		1.07	1.07		1.07	1.07		1.07	1.07		1.07	1.07	
35 LOCATION FROM HORIZONTAL REF DATUM INCHES															
36 PRESSURE JET & BLADE SECTION AREA FOR DUCT															
37 TIP JET THRUST		GEAR*** RATIO			GEAR*** RATIO			GEAR*** RATIO			GEAR*** RATIO			GEAR*** RATIO	
38 POWER TRANSMISSION DATA	HP	RPM		HP	RPM		HP	RPM		HP	RPM		HP	RPM	
39 MAX POWER - TAKE-OFF	1155.	40,000	117:1	1120	40,000	117:1	1034	40,000	117:1	1165	40,000	117:1	1292	40,000	117:1
40 LANDING GEAR TYPE-~~BICYCLE-TRICYCLE ARRANGEMENT~~-SKID ~~SKIRTED~~ MAIN-AFT AUX-FWD															
41 GEAR LENGTH - OLEO EXTND CL AXLE TO CL TRUNNION															
42 OLEO TRAVEL - FULL EXTENDED TO COMPRESSED INCHES															
43 WHEEL SIZE AND NUMBER REQUIRED															
44 FUEL AND OIL SYSTEM LOCATION NO. TANKS ***** GALS NO. TANKS ***** GALS															
45 CAPACITY PROTECTED															
46 FUEL - BUILT IN															
47 FUEL - EXTERNAL															
48 LUBRICATING SYSTEM															
49 HYDRAULIC SYSTEM															
50 STRUCTURAL DATA - CONDITION FUEL IN DESIGN STRESS															
51 WING-LB GROSS WT GROSS WT ULT LF	DESIGN GROSS WT	STRESS GROSS WT	ULT LF	DESIGN GROSS WT	STRESS GROSS WT	ULT LF	DESIGN GROSS WT	STRESS GROSS WT	ULT LF	DESIGN GROSS WT	STRESS GROSS WT	ULT LF	DESIGN GROSS WT	STRESS GROSS WT	ULT LF
52 FLIGHT 6660.	6525		5.25	6400		5.25	6400		5.25	6620		5.25	7050		5.25
53 LANDING															
54 % DESIGN LOAD WING % FWD RTR % AFT RTR %															
55															
56 TYPE OF POWER TRANSMISSION - GEARED - ~~PRESSURE JET~~ - ~~FAN JET~~															
57															

* PARALLEL TO CL OF ROTORCRAFT *** GEAR RATIO-ENG TO ROTOR
** CROSS OUT NON-APPLICABLE TYPE **** TOTAL USEABLE CAPACITY
***** REFER TO PARA. 3.1.1.3-ITEMS 6-33 & 6-34