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RESEARCH REPORT
JET-FLAP ROTOR
PRELIMINARY
APPLICATION STUDY
VOLUME I SUMMARY REPORT



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VOUGHT AERONAUTICS DIVISION
LTV AEROSPACE CORPORATION
P.O. BOX 5607 DALLAS, TEXAS 75222

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1.0 SUMMARY

Results of a Preliminary Application Study of the jet-flap rotor are presented and discussed. Because of its potential features in these areas, heavy lift and high speed applications were investigated. Comparison of the jet-flap vehicles with shaft-driven helicopters of comparable technology was made. Because of the limited technological base of the jet-flap rotor, uncertainties exist as to the achievable values of some of the design parameters. Therefore a number of side studies were conducted to determine the sensitivities of these parameters on the design. Minimum gross weight was the primary design criterion in the study, but some preliminary cost studies were also conducted.

It was found that the jet-flap rotor (and variation thereof) has the potential for significantly improving the aerodynamic characteristics of high speed rotary wing flight. However, power and fuel requirements to achieve these superior flying qualities are high. With ^{Certain} ~~assumptions~~ assumptions for design and performance parameters, jet-flap helicopters of significantly lower empty weight than that of comparable shaft-driven helicopters to perform equal missions can be visualized. ~~However,~~ ^{Since} the amount of fuel required by the jet-flap vehicle tends to offset the empty weight advantage ^{however} ~~since the fuel requirements~~ ^{which} ~~lead to high operating costs.~~ Because of the lack of specific design and operating data, no clear-cut total systems cost advantage of the jet-flap helicopter over the shaft-driven helicopter can be predicted for missions which the conventional helicopter can perform well, ~~although~~ ^{when} intangibles (with respect to cost and performance) such as high speed vibration, maintainability, and fatigue are considered, the high speed jet-flap offers potential advantages.

The effects on vehicle design characteristics of thrust recovery, duct loss, parasite drag, duct area ratio (denoting solidity), disc loading, tip speed, power split, and auxiliary thrust in high speed flight were investigated. Of these, the thrust recovery factor is most important as the effect of the others is either small or best values could be predicted with high confidence. Thrust recovery is difficult to determine experimentally, and in the centrifugal field with the rectangular nozzle, the uncertainty is aggravated. Gross weight variation for the high speed case can be from +14 to -9 percent for a thrust recovery range of 80 to 100 percent. For the short-range, heavy-lift case, the sensitivity is appreciably lower. This is attributable to lower fuel weight fraction.

This application study was parametric in nature, and many point designs were sized with different generalized input characteristics. The framework is not established as a departure point for more specific design studies and for studies of the "intangibles."

2.0 INTRODUCTION

This report presents results of a Preliminary Application Study exploring the potential capability of vehicles equipped with a jet-flap rotor in performing typical high speed and heavy-lift helicopter missions. It was conducted by the Vought Aeronautics Division of the LTV Aerospace Corporation (LTVAC) under contract to NASA Ames Research Center and the U. S. Army Aeronautical Research Laboratory, Moffett Field, California. Technical support was provided by personnel from Giravions-Dorand of France.

The study reported here complements analytical and experimental investigations conducted by Giravions-Dorand under contract to the U. S. Army and NASA Ames over approximately the last decade. Full-scale tests of a 39-foot-diameter rotor designated DH 2011 have been conducted in the Ames 40x80-Foot tunnel. Results of those tests and additional studies by NASA (TND-3028) revealed that large forces per unit blade area developed by the jet-flap rotor, exceed conventional rotor capabilities by factors of 2 and more. There was no evidence of retreating blade stall. The increased lift and propulsive force capabilities demonstrated, which did not represent jet-flap concept limits, indicate the jet-flap rotor to be both feasible and of potential value for high-speed rotors, low or moderate speed rotors where high lift is required, or for helicopters where high lift per unit area may be utilized to reduce the rotor size. The present study was initiated for the purpose of examining the jet-flap helicopter as a total aircraft system and compare its performance capability with that of a shaft-driven helicopter. Additional test of the DH 2011 rotor, scheduled to be performed in the Ames Research Center 40x80-foot tunnel in the spring of 1969, should further serve to experimentally demonstrate the performance potential of the jet-flap rotor under severe control conditions.

af In contrast to most application studies, the aircraft's mission was not specified or defined by the Statement of Work. Rather the contractor conducted a mission analysis in which high speed and heavy lift missions of all types were surveyed, and representative missions in each of the two categories were defined as base missions about which to conduct the parametric study.

Literally hundreds of vehicles were actually sized in the course of the study through the use of the computer programs. In formulating the vehicle concepts, emphasis was on the performance, while just enough design work was accomplished to provide an understanding of the essentials of this type of vehicle, particularly with regard to problem areas. Thus typical design concepts of the rotor, hub, and other mechanisms were formulated to provide the required credibility. No design optimizations were made but rather optimizations were limited to the rotor parameters which would yield minimum gross weight for the designated design missions.

One of the primary goals of the study was to determine the mission best suited to utilize the jet-flap rotor's capability and potential. Additionally, the study was to identify areas of research required to enhance the use of the jet-flap rotor concept.

In the first phase of the study, the jet-flap optimizations were based, as much as possible, on the DH 2011 test rotor concept. This is a cold cycle system, utilizing a turboshaft engine driving a compressor. Base point designs were established for each of the selected missions based on estimates of performance parameters judged to be achievable in the time period with a reasonable confidence level. Under these ground rules, for the

missions selected, results indicated that the jet-flap system did not possess the necessary degree of improvement in cost-effectiveness to present a salable approach.

It was evident, from some side studies conducted in the first phase, that significant improvement could be realized by utilizing a warm or hot propulsion cycle. Of interest also was the improvement that might be realizable if more optimistic values of parameters such as thrust recovery factor were assumed.

Thus, the next phase of the study emphasized the sensitivity of key parameters, particularly in the high speed application. Primary variables for the sensitivity study were specified as thrust recovery factor, propulsion system pressure loss, and parasite drag factor. Other factors whose sensitivity was examined in side studies included area ratio (ratio of duct area to blade profile area), pressure ratio, tip speed, diameter, (disc loading), and direct thrust compounding during high speed. Data are presented in such a way that the design characteristics can be estimated for any combinations of values within the range taken. The range of the parameters initially were arbitrarily selected even though it was known that some of the variables were not totally independent. Realistic values of the variables based on LTVAC technical judgment are indicated.

In this volume, a concise discussion of the methodology, results, and conclusions of the basic application study are presented. Additional data such as descriptions of computer data and substantiating data are reported in Volume II.

3.0 STUDY METHODS AND GROUND RULES

3.1 PROPULSION

The propulsion aspect of the jet-flap application study consists primarily of an assessment of the physical and thermodynamic characteristics of several types of jet-flap systems. Included are systems in which all propulsion is provided by the jet-flap, systems in which jet flap is combined with other types of propulsion systems, and a comparative conventional shaft-driven system.

A performance calculation methodology was developed, which was used to assess the individual systems, and to examine the effect of key propulsion parameters on system efficiency.

To assist in this endeavor, informal inquiries were submitted to several engine manufacturers regarding the performance and physical characteristics of the turbomachinery associated with the various systems. The results of these inquiries are indicated where appropriate. All engines selected for use in this study are judged to be of approximately equivalent technology level.

3.1.1 Jet Flap Propulsion System Requirements

The primary function of the jet-flap propulsion system is, of course, to provide main rotor power. Secondary functions include hover yaw control forces, aircraft accessory power, and in some cases, direct cruise thrust.

As in all pneumatic rotor drive systems, antitorque requirements are theoretically zero. Some force is required, however, to offset rotor downwash swirl and bearing friction, both of which tend to rotate the fuselage with the rotor. Should the efflux from the jet-flap nozzle impinge on the fuselage so as to produce a torque about the rotor mast, this force

will tend to rotate the fuselage in a direction opposite that of the rotor. In addition, a controllable force is required in both directions for hover and low speed heading control. An analysis of the net antitorque and control force required has not been made in this study. However, reasonable penalties have been assessed when appropriate, as indicated in the individual system descriptions.

3.1.2 Propulsion System Concepts

3.1.2.1 Shaft-Driven Rotor

Figure 3-1 shows the conventional mechanical drive system, which serves as a basis for comparison with the jet-flap systems in this study.

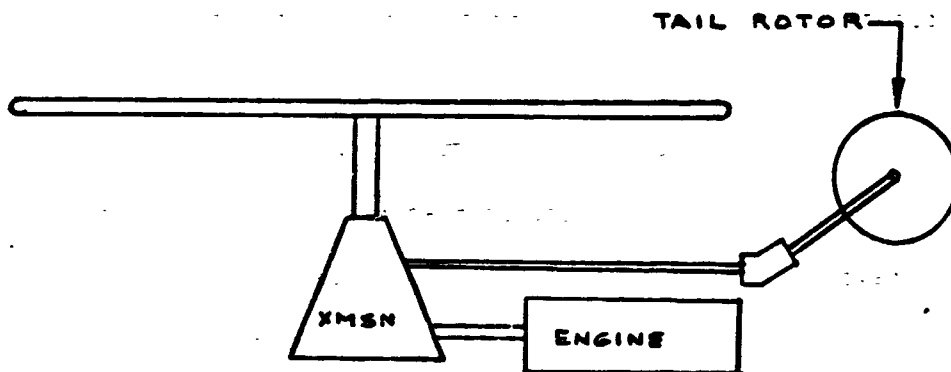


Figure 3-1 Mechanical Drive System

Turboshaft engine performance and scaling curves were based on advanced study data from Allison. Inlet duct recovery, P_{T_2}/P_{T_0} , was set at .990 in hover, increasing to .995 at velocities greater than 50 knots. Engine performance with this duct recovery was obtained from a computer routine supplied by the engine manufacturer. The following power losses were then applied so as to obtain net power available to the rotor:

Main transmission and cooling loss:	4% of input power
Accessory power extraction:	2% of input
Power to tail rotor:	7% of output to main rotor at hover, decreasing to 1.3% at 200 knots
Tail rotor gearboxes:	2% of tail rotor power

These losses combine to give an overall transmission efficiency (rotor horsepower/actual engine SHP) of 87.8 percent at hover, increasing to 92.7 percent at 200 knots or greater. Combined inlet loss and transmission losses result in a hover efficiency (rotor horsepower/rated engine SHP) of 86.2 percent. Rated power is the specification power at the "take-off" (10 minutes) power setting, sea level standard static conditions.

Propulsion system performance in terms of rotor horsepower available, engine fuel flow, engine residual net thrust, and engine physical size were calculated as a function of rated engine power, power setting, and airspeed.

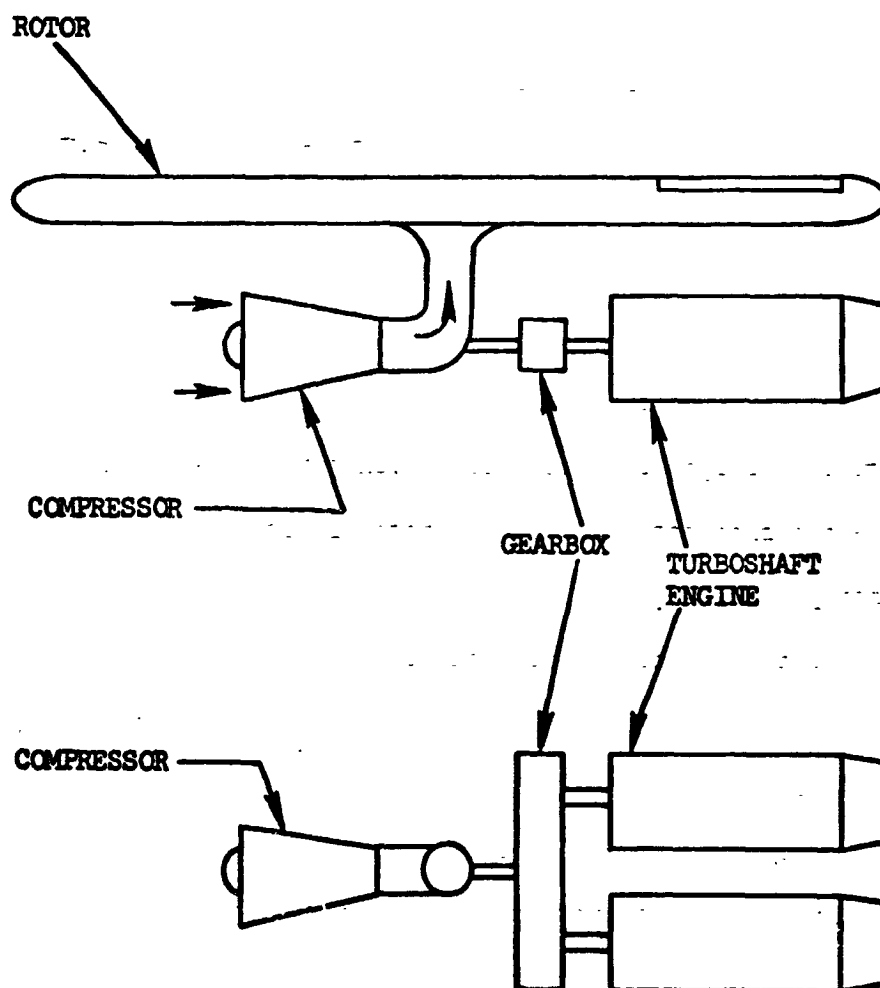
3.1.2.2 Jet-Flap Rotors

There are three basic jet-flap propulsion cycles, nominally categorized as "cold", "warm", and "hot," depending on the relative temperature of the gas which is delivered to the rotor. The characteristics of these three systems are described in the following paragraphs, together with those of several composite systems.

(1) Cold Cycle System

Figure 3-2 shows a typical cold cycle system. This is essentially the same as the system which was used in the full scale

COMPRESSOR DRIVE BY TURBOSHAFT ENGINES



PLAN VIEW~ROTOR REMOVED

Figure 3-2 Basic Cold Cycle System

jet-flap rotor tests.¹ Rotor gas consists of compressed air, supplied by a mechanical compressor. The compressor is driven by turboshaft engines, coupled to the compressor through a gearbox. The gearbox permits "engine out" operation, matches engine and compressor speed requirements, and provides for accessory power extraction. It is estimated that sufficient low speed directional control force can be obtained by vectoring the residual thrust of the turboshaft engines; therefore, no propulsion penalty is assessed. As flight speed increases, directional control is taken over by aerodynamic forces, and the engine exhausts are vectored aft.

The same turboshaft engine and inlet are used in the mechanical and cold cycle jet-flap systems. Therefore, the engine performance data generated for the mechanical rotor was used for the cold cycle jet-flap.

Compressor weight and size data were obtained from General Electric. Compressor performance maps were generated for a range of pressure ratios, based on in-house data and on information from NACA reports RM-E54G26A, RM-E54I31, RM-55A27, and RM-E56I07A. Figures 3-3, 3-4 and 3-5 show the resultant maps for design pressure ratios of 2.0, 3.0 and 4.2 respectively. Table 3-1 summarizes the compressor characteristics. Pressure ratio selection and detailed performance calculation procedures are discussed later in this section.

(2) Warm Cycle System

Figure 3-6A shows the warm cycle system. Turbofan engines, designed to permit mixing of the cold fan air with the hot turbine exhaust gas, are used to provide a "warm" gas mixture to the rotor. Diverter valves

¹ "DH 2011 Experimental Jet-Flap Rotor" Giravions Dorand Co. Document DH 2011-07-2197 PH 3/2, Volume 2.

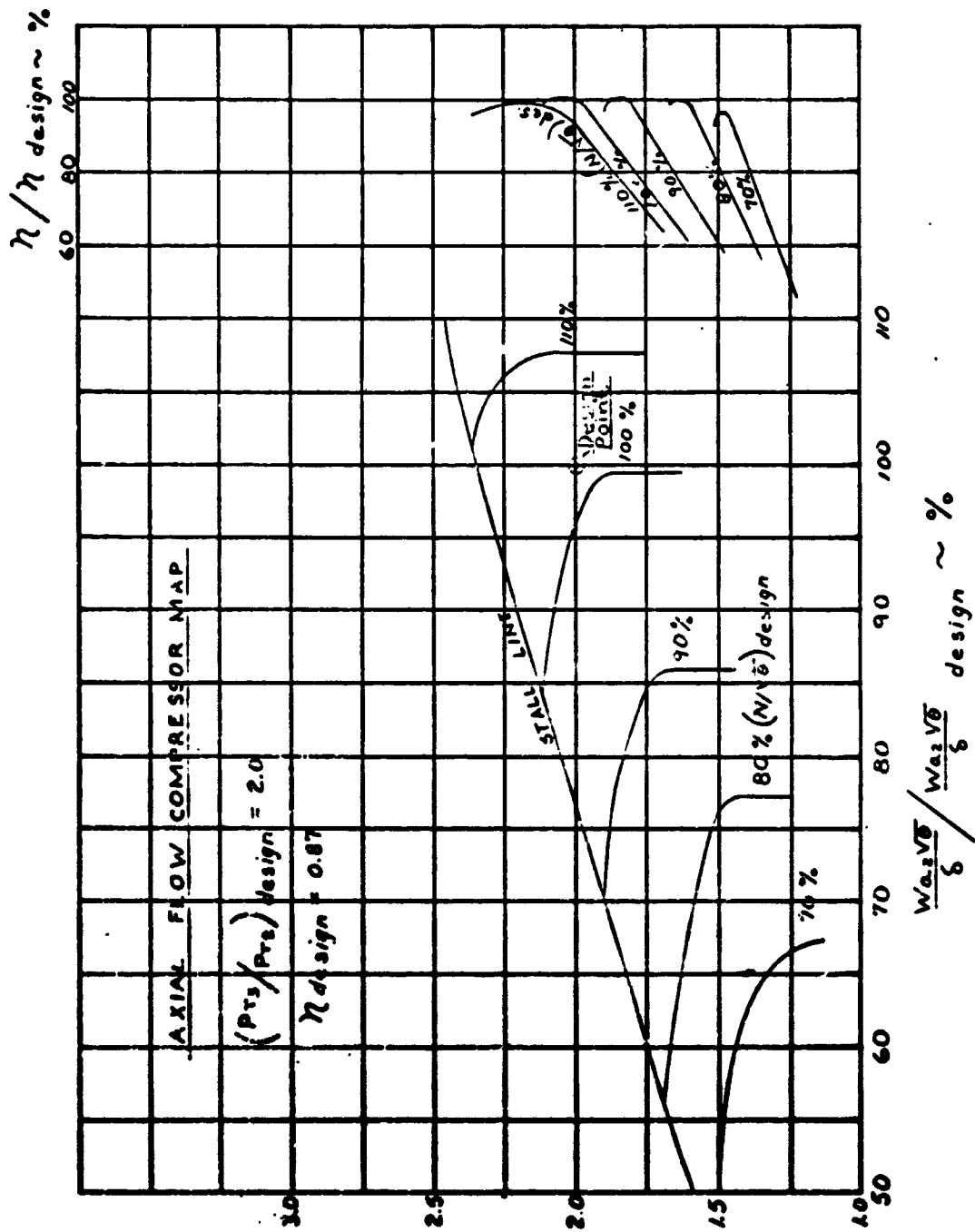
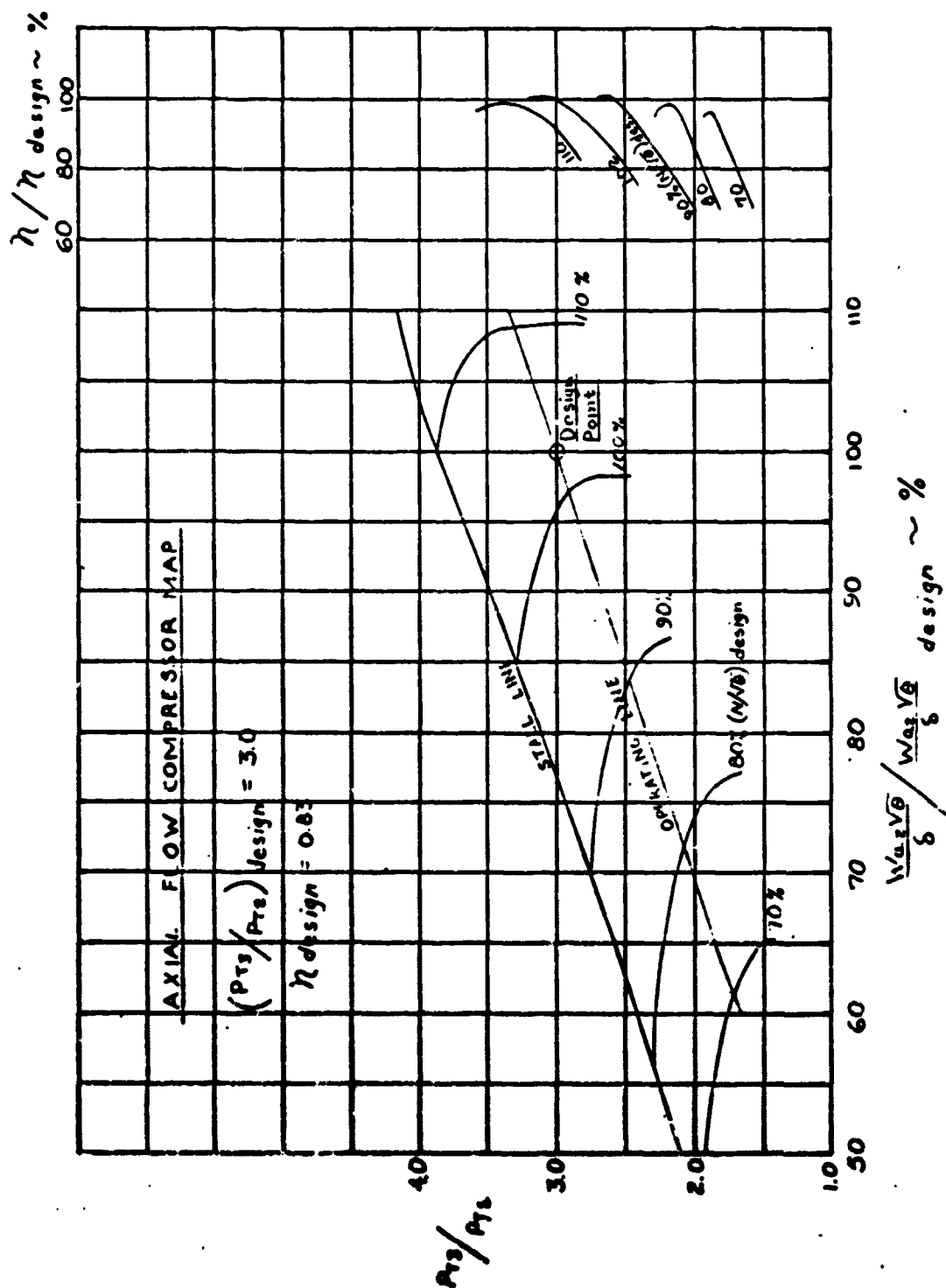


Figure 3.9

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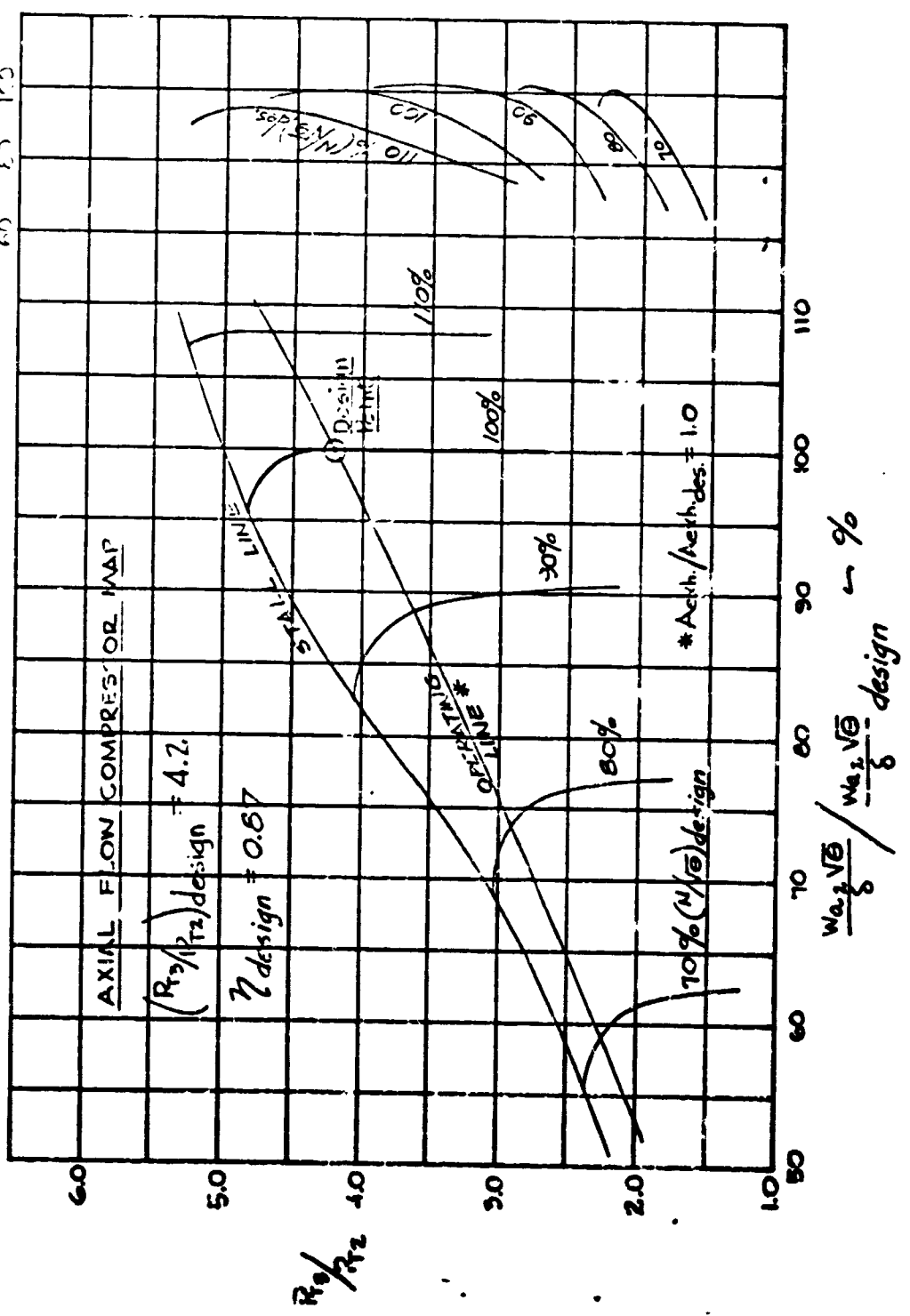


Figure 3-5 Axial Flow Compressor Map

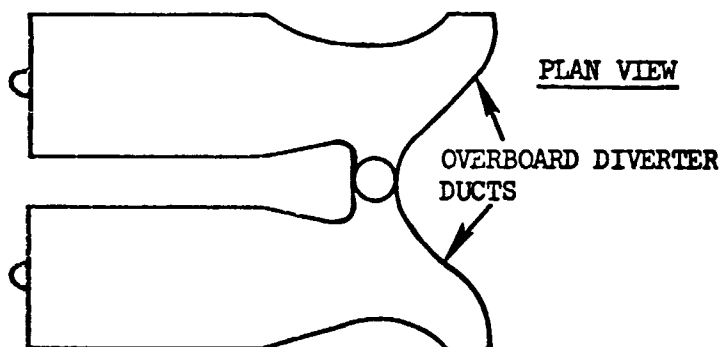
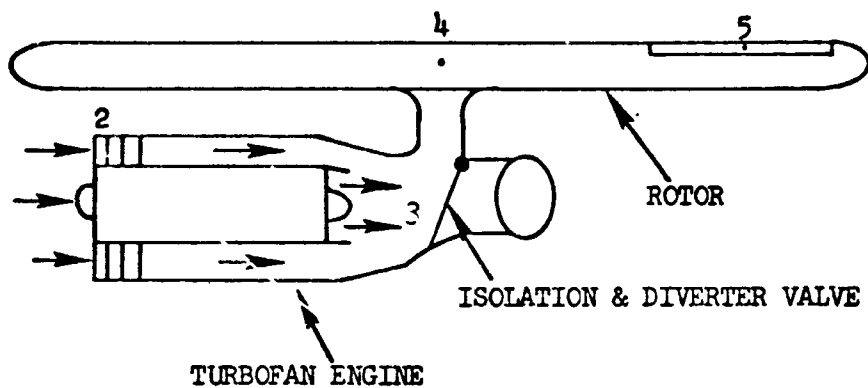
TABLE 3-1 COLD CYCLE COMPRESSOR CHARACTERISTICS
(BASED ON RATED SHP OF ENGINE = 10,000 HP)

PRESS. RATIO	P_{T_3} PSIA	T_{T_3} °F	lb/sec	$\frac{GHP_3}{HP}$	WEIGHT lb	$\frac{GHP_3}{SHP \text{ rated}}$ %	$\frac{GHP_3}{W_F}$ HP/lb/hr	$\frac{GHP_3}{HF}$ lb
2.0	29.1	139.3	210.6	8220	250	82.2	1.95	32.88
3.0	43.7	278.6	125.1	8391	190	83.9	1.99	44.16
4.2	61.1	360.9	91.0	8492	190	84.9	2.01	44.69

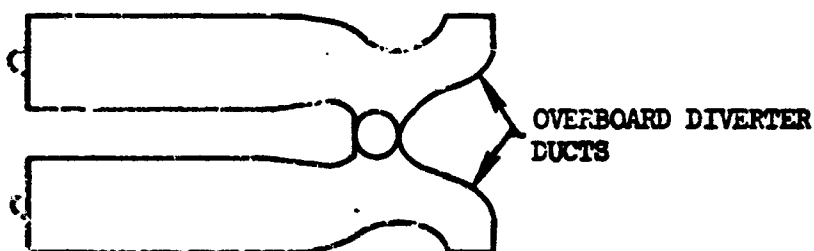
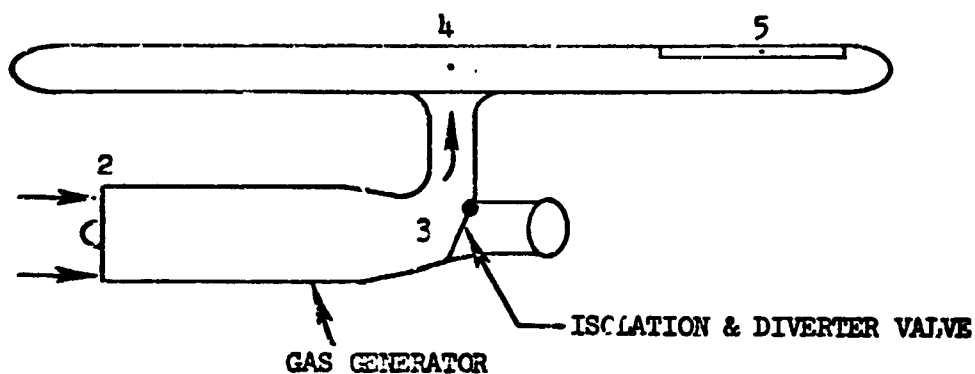
SHP rated = 10,000

Installed Engine Fuel Flow = 4223 lb/hr

= 87%



A. WARM CYCLE



B. HOT CYCLE

Figure 3-6 Warm and Hot Cycle Systems

and overboard ducts provide for engine starting, maintenance run up, and isolation of a failed engine. Directional control in hover is provided by control jets which use gas bled from the main rotor duct.

Specific warm cycle performance in this report is based on advanced turbofan study data provided by Pratt & Whitney. The characteristics of these engines, identified for convenience as "Engine 3" through "Engine 6", are shown by Table 3-2. These engines were selected so as to cover a range of mixed flow pressure ratios. Figure 3-7 shows the relationship between bypass ratio and pressure ratio for these engines. The significance of these effects will be subsequently covered. Yaw control and leakage flow is estimated at 5 percent of total flow in hover, decreasing to 2 percent as airspeed increases. This penalty is based on informal discussions with Dornier engineers on the performance of the DO-132 hot cycle helicopter.

(3) Hot Cycle System

Figure 3-6B shows the hot cycle system utilized in this study. Rotor gas in this case is supplied by gas generators, which are essentially turbojet engines. The ducting system, yaw control system, and flow penalties are the same as for the warm cycle system.

Specific hot cycle performance in this report is based on advanced gas generator study data provided by General Electric. Hot cycle engine characteristics, identified as "Engines 1 and 2", are shown by Table 3-2. Their relationship with the warm cycle engines on the pressure ratio scale is shown by Figure 3-7.

3.1.2.3 Composite Systems

The jet-flap system can be combined with other rotor-drive systems to form various composite systems. The objective of these systems

TABLE 3-2 HOT & WARM CYCLE ENGINE CHARACTERISTICS

ENG/ CYCLE	FN RATED LB.	PRESS RATIO PT3/PT2	BYPASS RATIO	PT3 PSIA	TT3 °F	WG3 lb/sec	GHP3 HP	WEIGHT lb	SFC lb/hr/lb	THRUST WEIGHT lb/lb	GHP3 WEIGHT HP/lb	GHP3 WF HP/lb/hr	GHP3 WG HP/lb/sec
1 HOT	6145	4.05	0	59.5	1574	70.0	16246	666	.929	9.23	24.4	2.85	232
2 HOT	5384	3.66	0	53.8	1420	66.0	13269	649	.882	8.30	20.4	2.79	201
3 WARM	9620	2.73	.80	40.1	847	158.0	17074	1295	.672	7.43	13.2	2.64	108
4 WARM	10200	2.42	1.15	35.6	727	189.0	16538	1410	.628	7.23	11.7	2.58	87.5
5 WARM	10900	2.21	1.50	32.5	634	220.0	16057	1455	.589	7.49	11.0	2.50	73.0
6 WARM	12000	1.91	2.20	28.1	515	281.0	15103	1635	.537	7.34	9.2	2.34	53.7

PRESSURE RATIO VS BYPASS RATIO
HOT & WARM CYCLE ENGINES

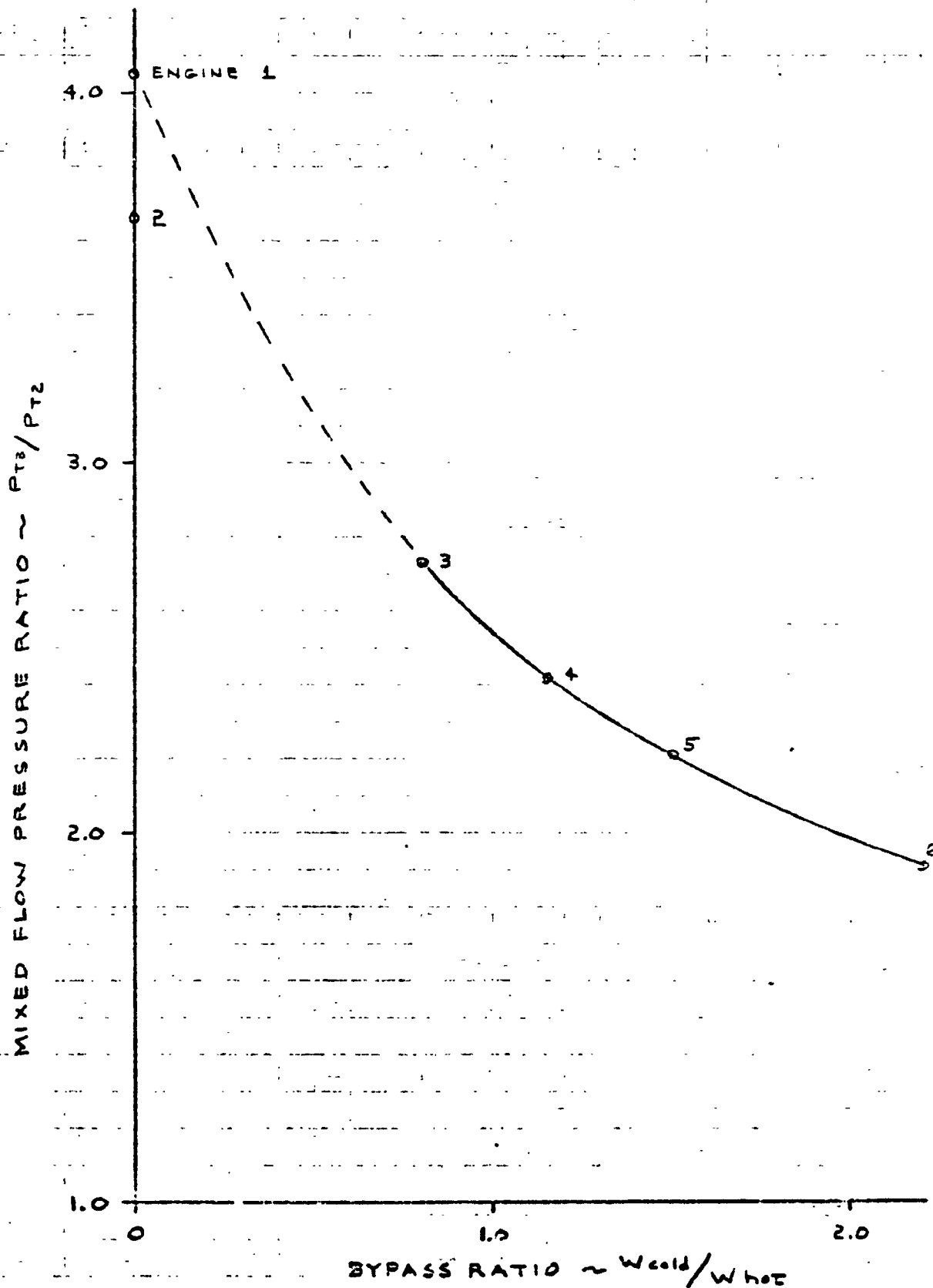


Figure 3-7

is to attain an overall efficiency higher than that of the pure jet-flap, while retaining most of its advantages. Composite systems which have been considered in this study are the partial mechanical, partial tip-jet, and partial direct propulsion (compound), described as follows:

(1) Partial Mechanical System

The partial mechanical system is essentially a conventional shaft-driven rotor system with jet-flap blowing superimposed. The system considered herein utilized the cold jet-flap cycle, with air supplied by a compressor driven by the main rotor transmission. A tail rotor is required to counteract the torque of the mechanical portion of the drive.

The turboshaft engine used in this system is the same as was used in the pure cold cycle system described previously.

(2) Partial Tip-Jet System

This is an all pneumatic system, in which the rotor air is apportioned between the jet-flap nozzles and the rotor tip-jet nozzles. The tip-jets are more efficient than the jet-flap because they are located at a greater effective radius, and are not deflected. They are purely propulsive, however, and do not contribute to rotor lift of control.

The partial tip-jet systems in this study utilize the hot cycle.

3.1.2.4 Compound Thrust Systems

The objective of these systems is to relieve the rotor of some of its forward thrust requirement by means of an auxiliary thrust producing device. A lower mission gross weight can be achieved if the efficiency of the auxiliary thruster exceeds that of the rotor by an amount sufficient to effect a fuel weight saving greater than the weight of the auxiliary thrust system. Obviously the required efficiency advantage is strongly influenced by the length of the mission.

Auxiliary thrust systems considered in this study are jet thrust, fan thrust, and propeller thrust.

(1) Jet-Thrust Compound System

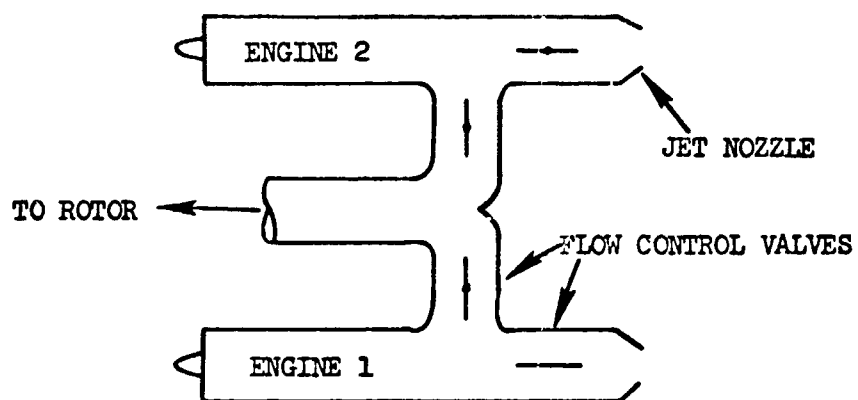
Jet-thrust compounding was considered in conjunction with the hot cycle jet-flap system as shown by Figure 3-8A. The basic hot cycle system has been modified by the addition of jet nozzles to which a portion of the gas generator flow can be directed in cruise. Variable area jet-flap and jet-thrust nozzles are required, in order for the engines to "see" the same equivalent exhaust system regardless of how the flow is prorated between the thrust nozzles and the rotor.

Although this system is relatively simple, the jet nozzles incur significant jet efflux losses within the speed capability of the helicopters considered.

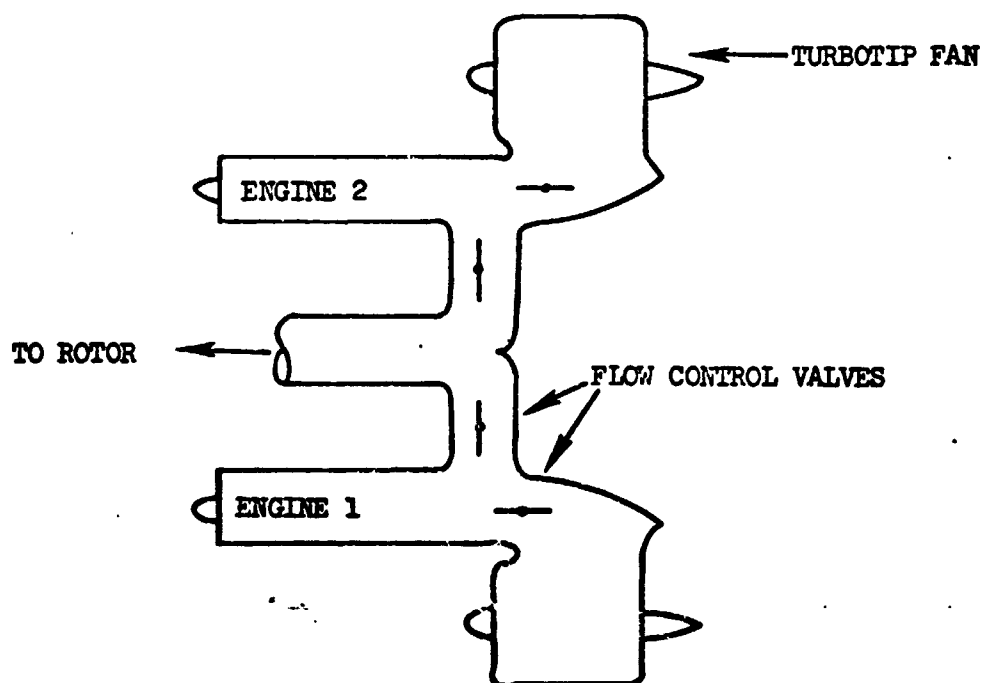
(2) Fan-Thrust Compound System

One method of fan-thrust compounding is to substitute turbofan engines for the gas generators in Figure 3-8A. This combines the simplicity and light weight of the jet-thrust compound system with the cruise efficiency of the fans. However, this method necessitates a warm cycle for the jet-flap rotor which has certain undesirable features, discussed subsequently in this section.

The hot cycle rotor can be combined with fan propulsion by the system shown in Figure 3-8B. In this case, a portion of the gas generator output is directed to tip turbine fans for cruise. Variable admission geometry is required on the fan turbine to provide for various flow splits. With this system, a relatively long mission is required in order for the fans to save their weight in fuel.



A. JET THRUST COMPOUNDED ~ HOT CYCLE



B. TURBOTIP FAN COMPOUNDED ~ HOT CYCLE

Figure 3-8

Both methods of fan-thrust compounding have been considered in this study. The tip turbine case is based on study data from General Electric.

(3) Turboprop Compound System

Of the several possible methods of turboprop compounding, the one selected for this study is simply to install a separate turboprop engine and propeller. This permits the propeller to be combined with any of the jet-flap cycles with a minimum of complication and technical risk.

3.1.2.5 Other Jet-Flap Systems

Numerous additional combinations, refinements, and rearrangements of jet-flap systems can be conjectured. Some which have been suggested include tip burning, swiveling tip nozzles, warm cycle with flow split (cold flaps plus hot tips), Cyclic flow rate variation, use of regenerators, and rotor-mounted engines.

3.1.3 Propulsion Parameters and Cycle Comparisons

3.1.3.1 ESHP

In the case of the conventional shaft-driven rotor, the propulsion system output is expressed in terms of the net power available at the rotor hub, downstream of all transmission and installation losses. Since the jet-flap rotor has no shaft power output, the power delivered to the rotor in this case is expressed as "equivalent" shaft horsepower, ESHP (not to be confused with turboprop engine terminology). ESHP is literally the equivalent, as far as the rotor is concerned, of the same amount of shaft power at the hub.

Analytically, ESHP is equal to the gross thrust horsepower of the jet ($M_J V_{J_{eff}} \bar{V}_T$), less the power returned to the rotor for centrifugal compression and Coriolis acceleration ($M_J \bar{V}_T^2$):

$$ESHP = \frac{M_J}{550} (V_{J_{eff}} - \bar{V}_T) \bar{V}_T$$

where: M_J = rotor mass flow, lb/sec/ft/sec²

$V_{J_{eff}}$ = effective nozzle velocity, ft/sec

$\bar{V}_T$ = rotor speed at center of jet, ft/sec

3.1.3.2 Thrust Recovery

Thrust recovery, as used herein, refers to the percentage of ideal nozzle thrust which is realized as a rotor torque producing force.

$$TR = \frac{F_{ideal}}{F_{actual}} = \frac{M_J V_{J_{ideal}}}{M_J V_{J_{eff}}}$$

This term is used in the determination of $V_{J_{eff}}$ in the ESHP equation.

This subject is subsequently covered in more detail in paragraph

3.1.6. In this study, the sensitivity of gross weight to thrust recovery values from 80 to 100 percent was examined.

3.1.3.3 Gas Horsepower

Gas horsepower refers to the energy potential which the gas possesses by virtue of its temperature, pressure and flow rate:

$$GHP = \frac{W_g \text{ lb/sec} \times C_p \text{ ft-lb/lb-}^\circ\text{R} \times T_T \text{ }^\circ\text{R}}{550 \text{ ft-lb/sec-HP}} \left[1 - \left(\frac{P_{AM}}{P_T} \right)^{\gamma-1/\gamma} \right]$$

Gas horsepower can be thought of as the shaft power which would be developed by an ideal turbine, if the gas were expanded through it to ambient pressure, with zero kinetic energy exhaust.

3.1.3.4 Pressure Drop

System pressure drop, as used herein, refers to the loss in total pressure between the gas generator outlet (Sta. 3) and the rotor nozzle (Sta. 5), expressed as a percentage of the pressure available:

$$\Delta P_{T_{3-5}} / P_{T_3}$$

Centrifugal compression is calculated separately and is not included in the ΔP figure. This subject is covered in detail in Paragraph 3.1.5.

3.1.3.5 Duct Mach Number

The design duct Mach number selection is a compromise between the flow area required to achieve a low Mach number, and the high pressure losses associated with a high Mach number. In a system with high pressure loss coefficients, the area increase required along the flow path to hold the Mach number constant can defeat the advantage of a low entry area if too high a Mach number is selected.

A duct Mach number optimization was not included in the present study; however, a value of 0.3 was selected based on past experience.

3.1.3.6 Rotor Propulsion System Efficiency

This parameter shows how effectively the gas horsepower delivered to the rotor is converted into equivalent shaft horsepower:

$$\eta_P = \text{ESHP} / \text{GHP}_3$$

Included are pressure drop, temperature drop, thrust recovery and energy lost in the jet efflux. In the ideal system, internal losses are zero and thrust recovery is 100 percent. In this case, the only loss is the energy remaining in the jet efflux, $\frac{M_J}{2} (V_J - \bar{V}_T)^2$.

$$\begin{aligned}
 \text{Therefore } \eta_{P_{ideal}} &= \frac{\text{ESHP}}{\text{ESHP} + \text{Jet Efflux Loss}} \\
 &= \frac{M_J (V_J - \bar{V}_T) \bar{V}_T}{M_J (V_J - \bar{V}_T) \bar{V}_T + \frac{M_J}{2} (V_J - \bar{V}_T)^2} \\
 &= \frac{\bar{V}_T}{\bar{V}_T + \frac{1}{2} (V_J - \bar{V}_T)} = \frac{2\bar{V}_T}{2\bar{V}_T + V_J - \bar{V}_T} \\
 &= \frac{2}{1 + \frac{V_J}{\bar{V}_T}}
 \end{aligned}$$

Therefore, for a given rotor tip speed, propulsion efficiency is improved by decreasing the jet velocity, which is effected by decreasing the gas temperature or pressure or both.

In order to deliver a certain level of thrust, as the jet velocity decreases, the mass flow must increase. In the limiting case, where $V_J = \bar{V}_T$, propulsive efficiency is 100 percent, but an infinite flow rate is required. The physical significance of this parameter is illustrated by comparing a lift jet VTOL aircraft with a helicopter of the same gross weight. The lift jets develop thrust by means of a low mass flow, high velocity jet. Downwash erosion effects are indicative of the wasted energy in the jet, leading to poor propulsive efficiency. The helicopter rotor develops lift by adding a small incremental velocity to a much larger mass of air, resulting in high propulsive efficiency.

Figure 3-9 shows the ideal rotor system propulsive efficiency of several of the jet-flap systems considered in this study.

3.1.3.7 Gas Generator Efficiency

This parameter shows the efficiency of the gas generator system, in terms of GHP delivered per pound of fuel consumed, as shown by Figure 3-10 for the various systems considered. It is interesting to note that whereas the low pressure cold cycle system is a clear choice on the basis of Figure 3-9, it is the least attractive in Figure 3-10. It appears that the cold cycle efficiently utilizes the gas delivered to the rotor, but generates the gas in an inefficient manner. The basic reason for the poor showing of the cold cycle as a gas generator is that an excessive number of energy transformations is required. The gas generator portion of the turboshaft engine converts fuel into a gas horsepower input to the power turbine, which converts the energy to shaft power. This power is delivered through a gearbox to a compressor, where it is converted back to gas power. Each of these transformations incurs losses.

In the warm cycle, only the cold portion of the flow goes through the intermediate shaft power link, which decreases with bypass ratio.

In the hot cycle, all of the energy is delivered in GHP form, resulting in the most efficient gas generation system.

3.1.3.8 Overall System Efficiency

The overall system efficiency combines gas generator efficiency and the rotor system efficiency, giving :

$$\eta_{\text{overall}} = \frac{\text{ESHP.}}{W_f \cdot \text{lb/hr} \times 7.27 \text{ HP/lb-hr}}$$

IDEAL PROPULSIVE EFFICIENCY

$$\Delta P_T / P_T = 0$$

$$\text{THRUST RECOV.} = 100\%$$

$$\eta_p = 2 / (1 + V_0 / \bar{V}_T) = \text{ESHP} / \text{GHP}_3$$

$$V_T = 640 \text{ FPS} \quad \bar{V}_T = 544 \text{ FPS}$$

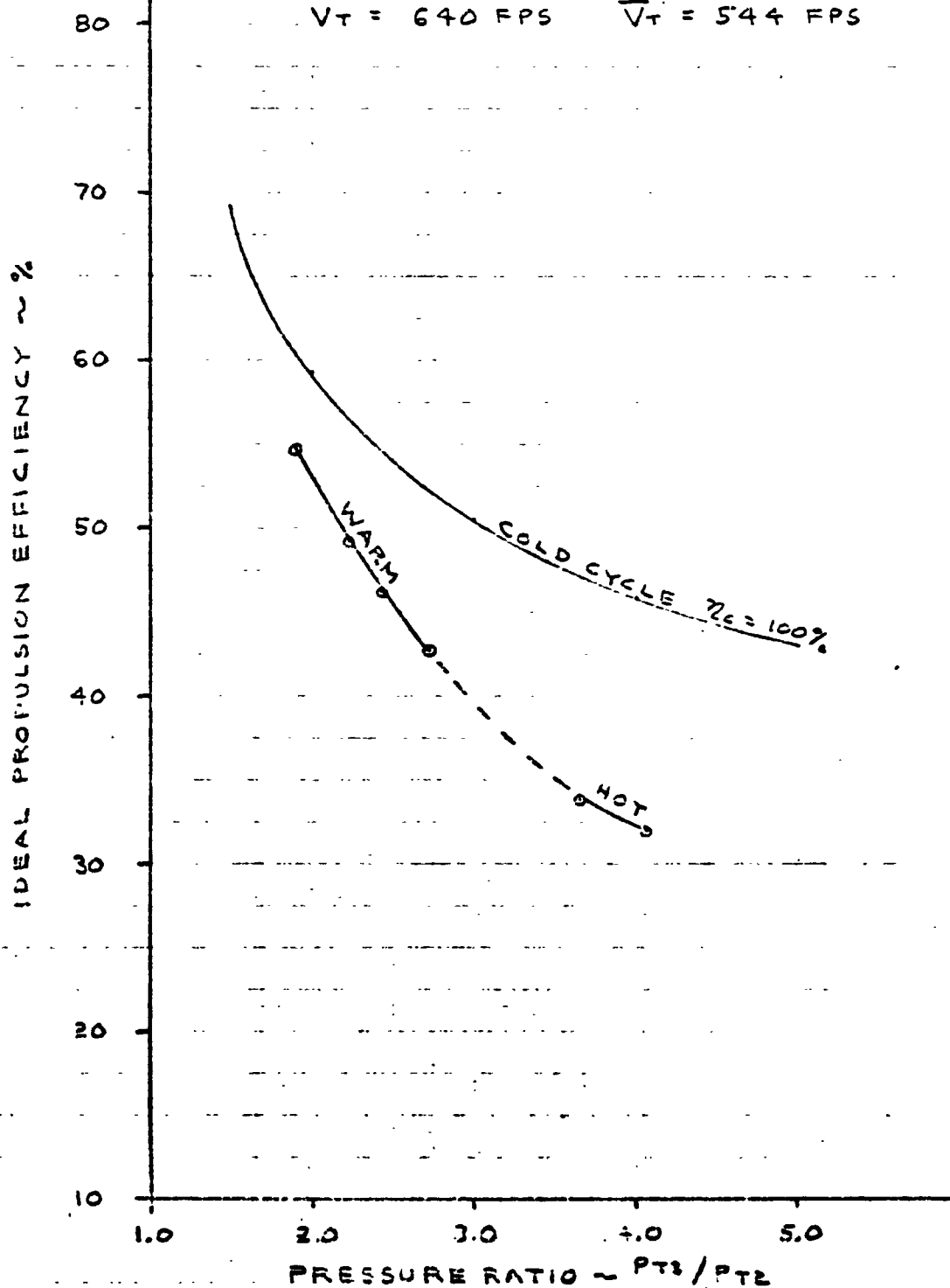


Figure 3.9

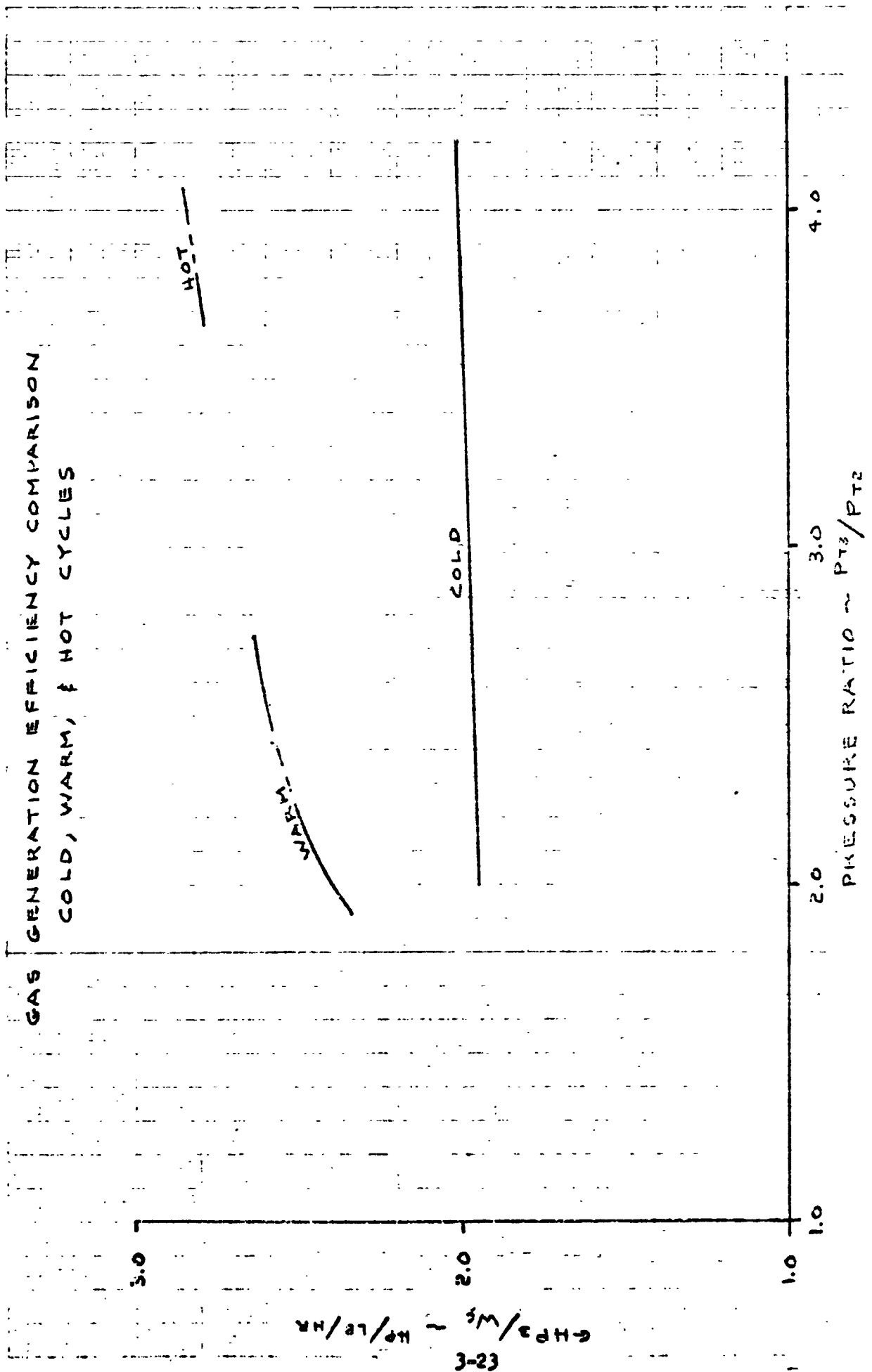


Figure 3-10

Figure 3-11 shows a measure of this efficiency in terms of $ESHP/W_p$ for the various systems, as a function of system losses. It can be noted that on an overall basis, the cycles are much more equivalent than Figures 3-9 or 3-10 would indicate. Furthermore, when realistic system losses are included, all of the systems and pressure ratios fall in a relatively narrow efficiency band. This is because system losses have a more pronounced effect on the low pressure systems, thus reducing their efficiency advantage.

3.1.3.9 Duct Area Required

The duct area required is a function of system efficiency, type of cycle, and design duct Mach number. It is expressed in terms of average total cross-sectional area required per ESHP delivered. The area required is calculated on the basis of average gas density in the ducting system. In order to hold a constant Mach number, the entry area will be somewhat smaller than the average value, and the area at the entrance to the rotor somewhat larger, due to decreased density as a result of pressure losses (Fanno line conditions). The area along the rotor will be essentially constant, due to centrifugal compression offsetting the effects of duct friction, as noted by Henry¹.

Figure 3-12 shows the flow area required for the various systems. The lower specific energy level of the low pressure end of the scale results in higher gas flow rates with consequent increase in duct area.

¹ "One-Dimensional, Compressible, Viscous Flow Relations Applicable to Flow in a Duct Helicopter Rotor", J. R. Henry, NACA TN 3089, 1953

OVERALL EFFICIENCY COMPARISON
S.L. STD. STATIC TAKEOFF POWER
VTIP = 640 FPS

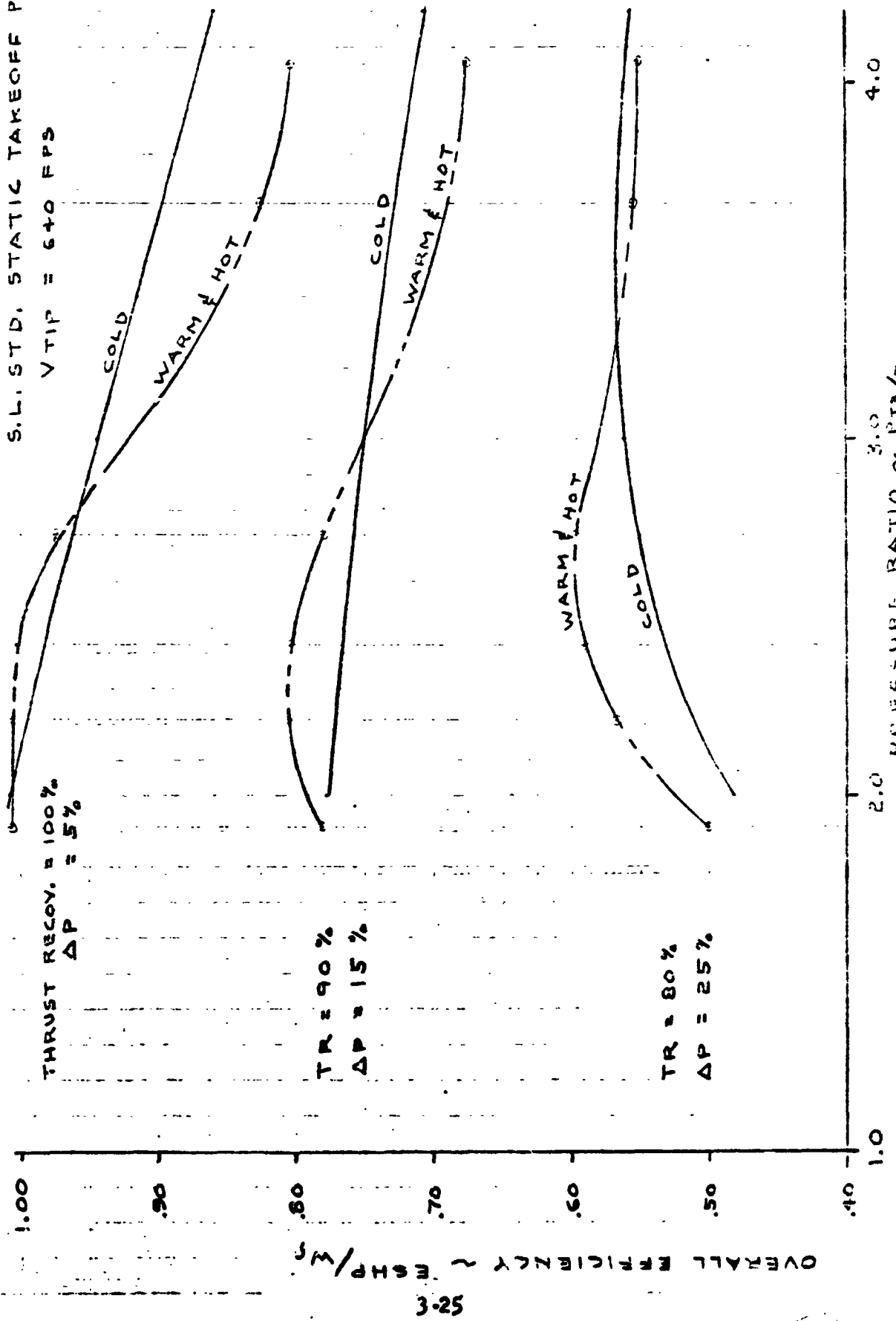


Figure 3-11

DUCT AREA REQUIRED
COLD, WARM & HOT CYCLE SYSTEMS
S.L. STD STATIC T/O POWER
VTIP = 640 FPS

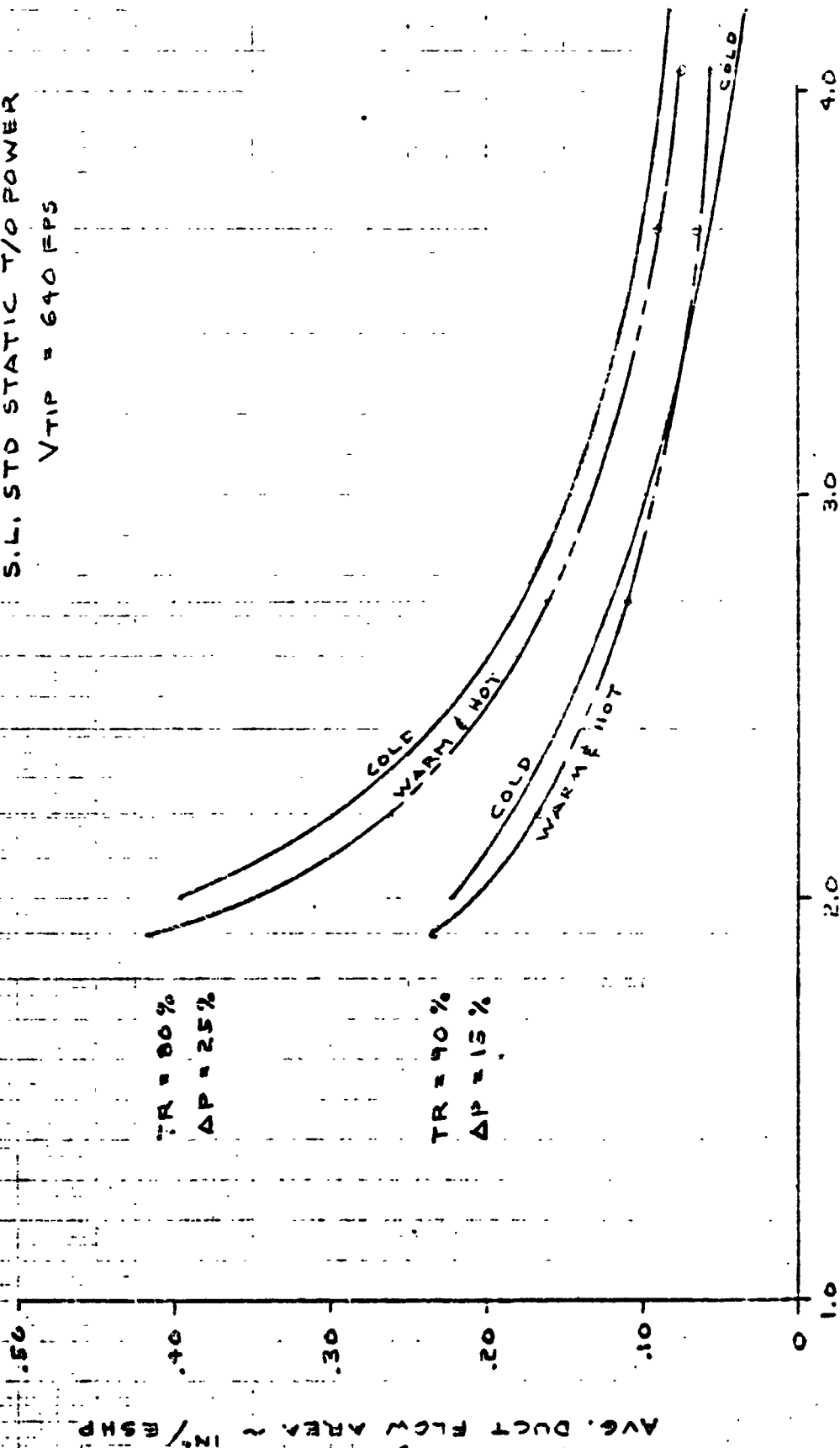


Figure 3-12
PRESSURE RATIO ~ P_1/P_2

3.1.3.10 Ram Drag

The ram drag associated with taking the rotor airflow on board is a function of the rotor airflow requirement and airplane velocity. For those systems using turboshaft engines, the ram drag of the engine airflow is included in the residual net thrust of the engine. Ram drag is a major deterrent to the use of low pressure, high flow cycles, especially in the high speed application. Although the incoming air contributes energy to the system in the amount of $M_o V_o^2/2$, this does not compensate for the resultant ram drag horsepower required, $M_J V_o \times V_o$.

Figure 3-13 and 3-14 show typical values of ram drag in high speed cruise.

3.1.3.11 Overall System Comparison

It is evident from the preceeding discussion that a propulsion cycle and pressure ratio cannot be selected on the basis of any one propulsion parameter. Propulsive efficiency alone is never used as the basis of choosing the system. Overall SFC and vehicle gross weight required to accomplish a specific mission are the predominant factors. Therefore, the final system selection was made by comparing the resultant gross weight of aircraft designed for the same mission, using different propulsion cycles. Figure 3-15 shows the results of this comparison, the mechanics of which are discussed in subsequent sections of this report. It is concluded that when all effects are considered, the hot cycle system results in the lightest weight vehicle. Other considerations such as technical risk or availability of components would affect the actual system selection for a specific application.

DRIVEN COMPRESSOR RAM DRAG
COLD CYCLE
NORMAL RATED POWER
S.L. STD.

08-3
RAM DRAG / SHP rated = lb/hp

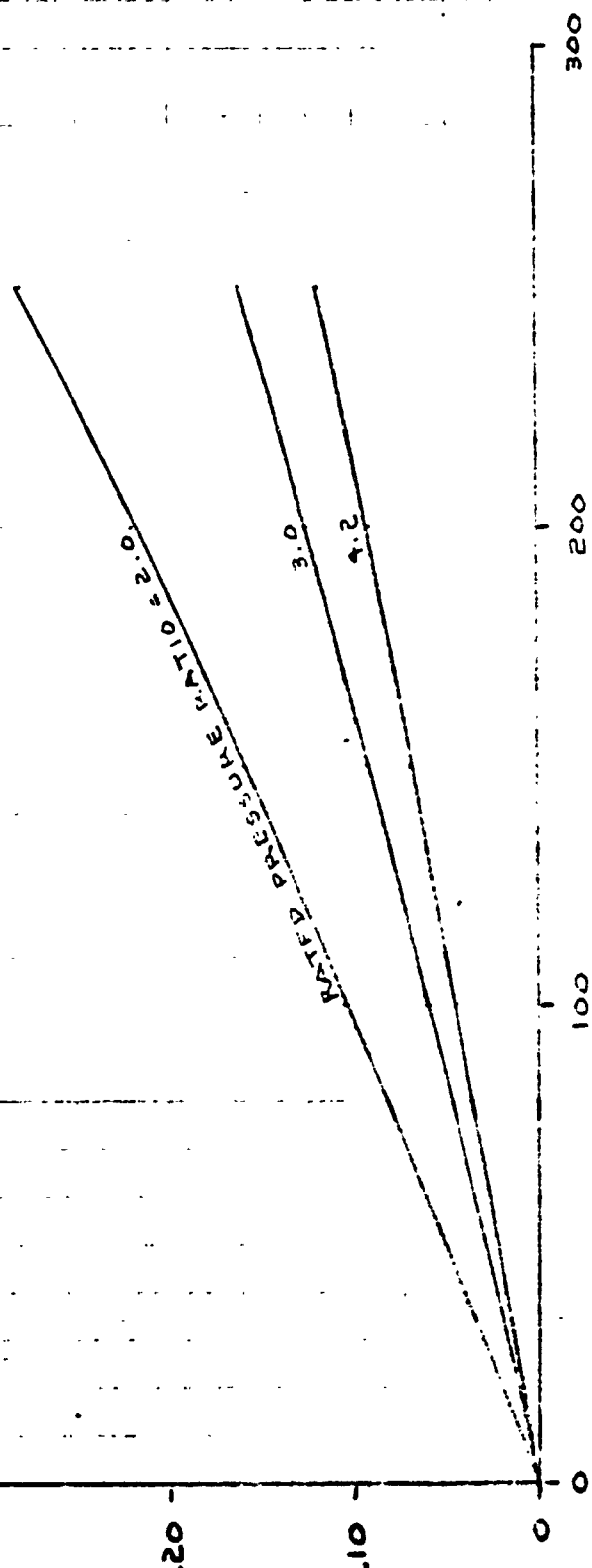


Figure 3-13

"REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR."

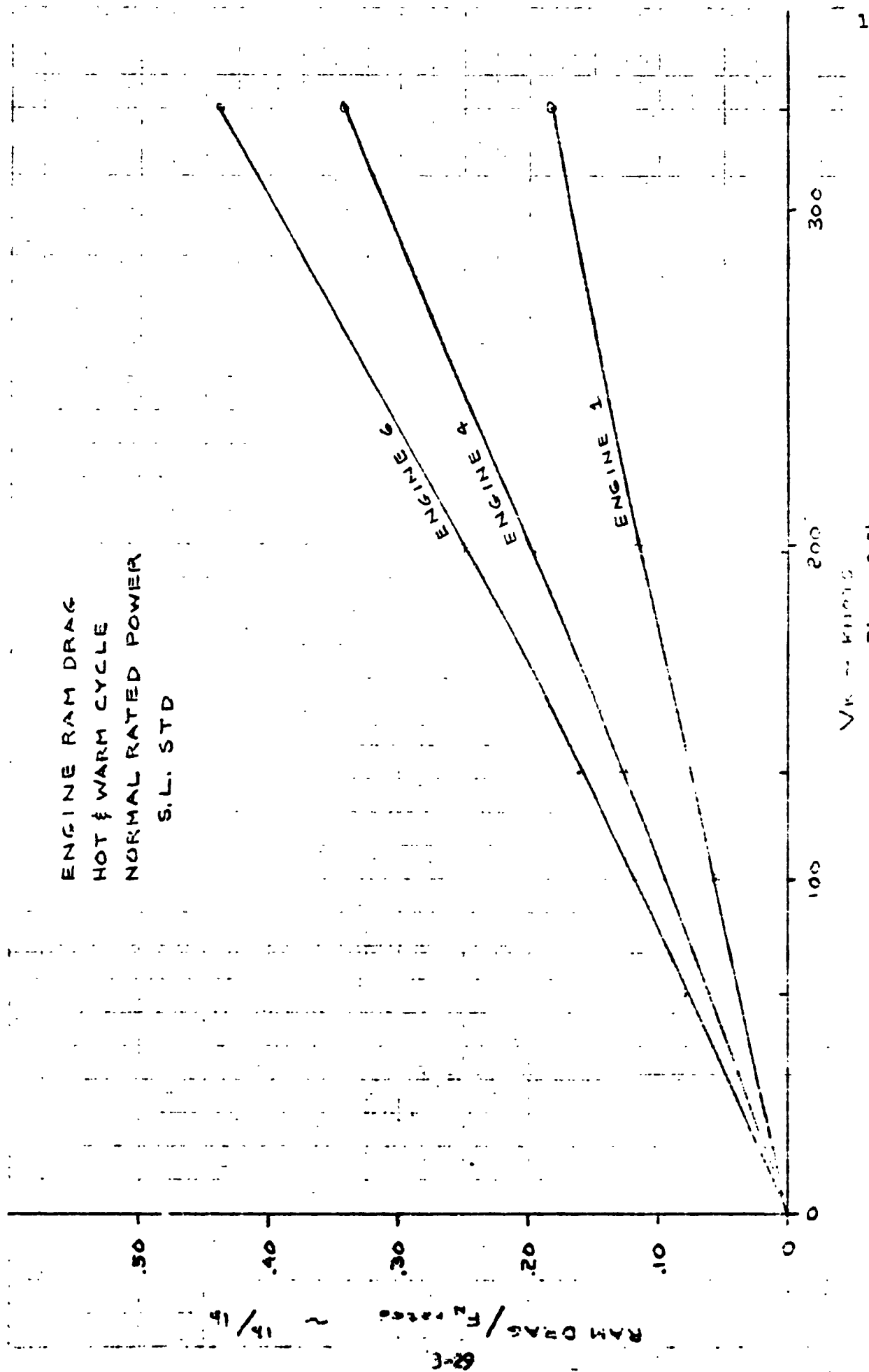


Figure 3-14

TAKEOFF GROSS WEIGHT vs. PRESSURE RATIO

TR=90%
ΔP=15%

DIA = 72.2 FT
 $V_T = 700 \text{ f/s}$

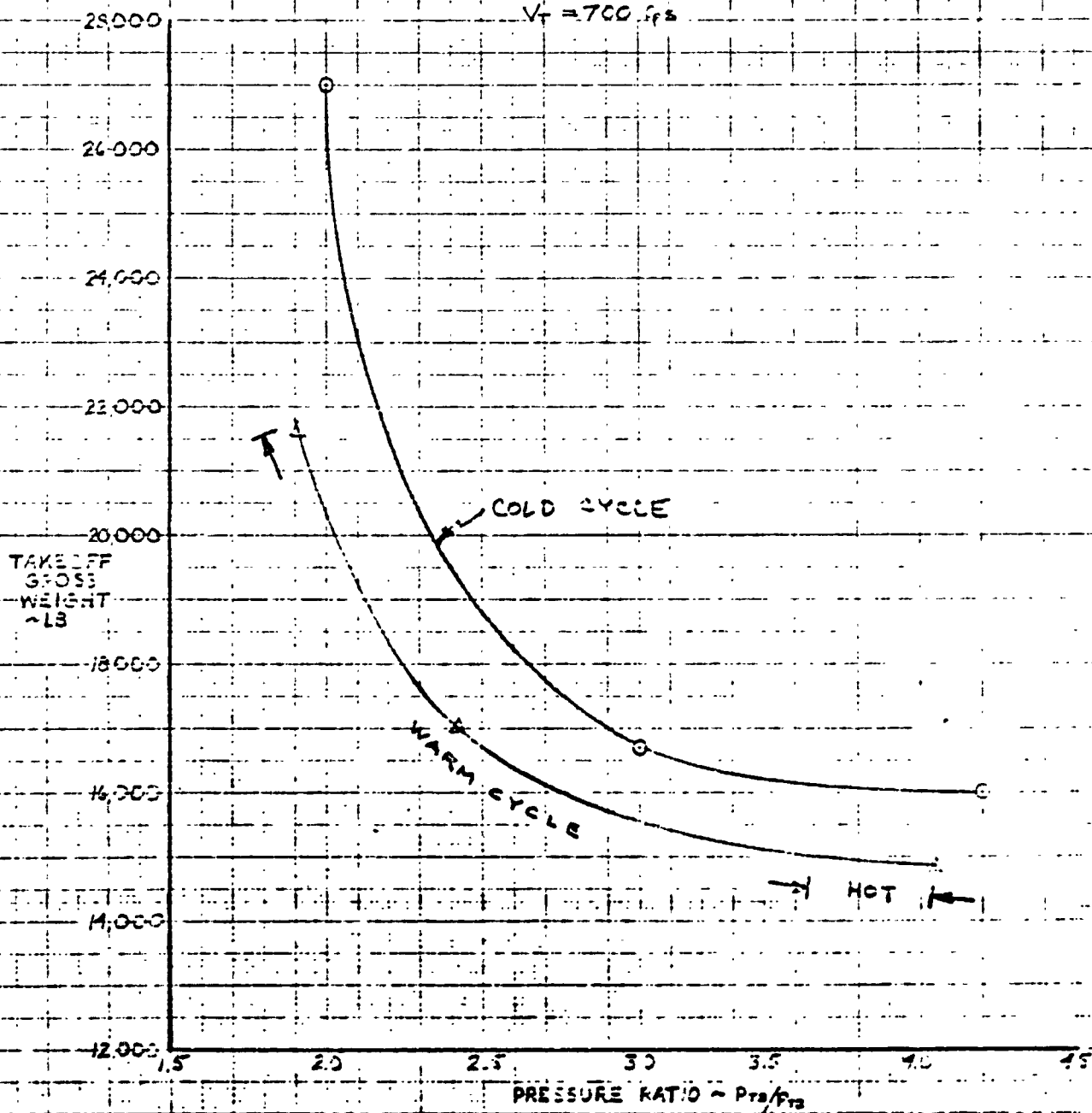


Figure 3-15

3.1.4 Propulsion System Performance Methodology

3.1.4.1 Cold Cycle

Engine and compressor performance were combined to obtain the gas temperature, pressure, and flow rate supplied to the rotor system, for a range of engine power settings and airplane velocities. The ESHP delivered to the rotor was then calculated for each of these conditions. The general calculation procedure is as follows:

- (1) Compressor airflow is related to power input by means of:

$$\frac{\text{SHP}}{\dot{W}_a} = \frac{C_p T_{T2}}{\eta_c} \left[\left(\frac{P_{T3}}{P_{T2}} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right]$$

From this relation, a curve of percent corrected airflow versus percent SHP is calculated, where 100% is defined as take-off power and $V = 0$.

- (2) For each power setting and airspeed condition, the percent power available is calculated and is applied to the compressor exit.
- (3) System pressure loss is subtracted to obtain pressure at point 4. Temperature drop from 3 to 4 is assumed as 5°F. (Stations per Figure 3-2)
- (4) Pressure and temperature increase in the rotor due to centrifugal compression are added algebraically with system losses in the rotor to obtain the net gas properties at the nozzle, Station 5. Rotor temperature drop due to heat loss from the duct is assumed to be 5°F.

- (5) Nozzle jet velocity is calculated from the gas conditions at Station 5.

- (6) Gross thrust horsepower is calculated from airflow and nozzle jet velocity. Thrust recovery factors are applied at this time.

$$GTHP = W_a/g \times (TR \times V_j) \bar{V}_t/550$$

- (7) Coriolis power is calculated from tip speed and airflow. This is the power required to compress the gas in the rotor, and to increase the angular momentum of the flow from zero (at the hub) to the value at the tip.

$$HP_{cor} = W_a/g \bar{V}_t^2/550$$

- (8) ESHP is obtained for each thrust recovery by subtracting HP Coriolis from the appropriate value of gross thrust horsepower.

$$ESHP = W_a/g (TR \times V_j - \bar{V}_t) \bar{V}_t/550$$

- (9) Compressor ram drag is calculated from airflow and airplane velocity.

$$D_r = W_a/g \times V_o$$

- (10) Average total duct area required is calculated based on average gas conditions and the Mach number function

$$(W_a \sqrt{T_m/P_m}) \text{ for Mach } 0.30.$$

Rotor ESHP, engine fuel flow, net thrust, and ram drag are then scaled as a function of rated engine power. Steps 1 through 10 were calculated by means of a computer routine, the details of which are contained in Volume II. Table 3-3 summarizes the resultant cold cycle system characteristics.

TABLE 3-3 PROPULSION SYSTEM CHARACTERISTICS COLD CYCLE

TAKEOFF POWER, $V_{TLP} = 640$ FPS

PRESS RATIO	A P	THRUST RECOV.	S.L. STD		6000 FT., 95°F		
			$\frac{A_{TOTAL}}{ESHP}$	$\frac{ESHP}{W}$	$\frac{ESHP}{SHP_{rated}}$	$\frac{A_{TOTAL}}{ESHP}$	$\frac{ESHP}{SHP_{rated}}$
	%	%	IN ² /HP	HP/LB/HR		IN ² /HP	
2.0	5	80	.2232	.692	.2923	.3135	.2061
		90	.1745	.881	.3719	.2455	.2632
		100	.1445	1.069	.4516	.2018	.3203
	15	80	.2874	.596	.2516	.4120	.1739
		90	.2217	.772	.3262	.3157	.2270
		100	.1804	.949	.4008	.2559	.2800
	25	80	.3954	.483	.2039	.5888	.1357
		90	.2959	.645	.2725	.4343	.1840
		100	.2364	.808	.3411	.3440	.2323
3.0	5	80	.1005	.658	.2778	.1444	.2492
		90	.0826	.801	.3382	.1184	.3039
		100	.0701	.944	.3535	.1003	.3587
	15	80	.1196	.513	.2589	.1734	.2301
		90	.0977	.750	.3168	.1413	.2825
		100	.0826	.887	.3748	.1192	.3348
	25	80	.1456	.562	.2372	.2138	.2083
		90	.1181	.693	.2925	.1726	.2579
		100	.0993	.823	.3477	.1448	.3075
4.2	5	80	.0593	.617	.2607	.0835	.1855
		90	.0495	.739	.3119	.0697	.2221
		100	.0426	.860	.3632	.0598	.2588
	15	80	.0689	.589	.2488	.0974	.1763
		90	.0574	.707	.2985	.0811	.2118
		100	.0492	.825	.3483	.0695	.2473
	25	80	.0836	.557	.2354	.1155	.1695
		90	.0675	.671	.2835	.0958	.2001
		100	.0577	.785	.3315	.0818	.2342

3.1.4.2 Warm and Hot Cycle

The warm and hot cycle are analyzed in the same manner, the difference being in the bypass ratio of the engines. The calculation procedure is somewhat more simple than that of the cold cycle, since the engine output is applied directly to the rotor, without going through the compressor link.

Engine exhaust gas temperature, pressure, and flow rate were obtained from manufacturer's data, using the inlet duct losses noted previously. Yaw control and leakage flow was deducted as noted previously. Specification engine fuel flow was increased 5 percent to account for accessory power extraction. Once the gas conditions are established, calculation of rotor system performance is the same as for the cold cycle. Actual calculations were performed using the computer routine detailed in Volume II.

Table 3-4 summarizes the characteristics of the warm and hot cycle.

3.1.5 Propulsion System Pressure Drop Analysis

Thus far, pressure drop has been discussed in terms of arbitrarily assumed values covering the probable realistic range. A pressure drop study was conducted to more narrowly define this probable range.

$$\Delta P_t / P_t = K_t \frac{\gamma M^2}{2 + \gamma M^2}$$

$\gamma = 1.4$ for cold cycle

$\gamma = 1.36$ for warm cycle

$\gamma = 1.32$ for hot cycle

$M = \text{Mach number of gas flow} = 0.30$

TABLE 3-4 PROPULSION SYSTEM CHARACTERISTICS
HOT & WARM CYCLE - ENGINES 1,2, & 3, - $V_{TIP} = 640$ FPS

ENGINE NO.	ΔP_{3-5} PT ₃	THRUST RECOV.	S.L. STD T/O POWER			6000 ft. 95° T/O POWER	
			$\frac{A_{TOTAL}}{ESHP}$	$\frac{ESHP}{Wf}$	$\frac{ESHP}{FN_{rated}}$	$\frac{A_{TOTAL}}{ESHP}$	$\frac{ESHP}{FN_{rated}}$
			IN^2/HP	$\text{HP}/\text{LB}/\text{HR}$	HP/LB	IN^2/HP	HP/LB
1	.5	.80	.0557	.6053	.5834	.0747	.4352
		.90	.0479	.7044	.6789	.0641	.5066
		1.00	.0420	.8035	.7744	.0562	.5779
	.15	.80	.0646	.5795	.5585	.0868	.4157
		.90	.0554	.6754	.6509	.0744	.4846
		1.00	.0454	.7713	.7433	.0652	.5535
	.25	.80	.0760	.5504	.5305	.1024	.3937
		.90	.0651	.6427	.6194	.0876	.4598
		1.00	.0569	.7349	.7083	.0766	.5260
2	.5	.80	.0650	.6172	.5718	.0942	.3945
		.90	.0557	.7206	.6676	.0806	.4612
		1.00	.0487	.8241	.7634	.0704	.5280
	.15	.80	.0758	.5878	.5445	.1100	.3735
		.90	.0648	.6875	.6369	.0939	.4376
		1.00	.0566	.7873	.7293	.0819	.5018
	.25	.80	.0897	.5546	.5137	.1318	.3497
		.90	.0765	.6502	.6023	.1122	.4109
		1.00	.0667	.7457	.6908	.0976	.4720
3	.5	.80	.1092	.7051	.4975	.1529	.3554
		.90	.0918	.8385	.5917	.1283	.4236
		1.00	.0792	.9719	.6858	.1105	.4919
	.15	.80	.1306	.6544	.4618	.1835	.3286
		.90	.1094	.7815	.5514	.1533	.3935
		1.00	.0941	.9086	.6411	.1316	.4584
	.25	.80	.1601	.5962	.4207	.2261	.2978
		.90	.1333	.7160	.5052	.1877	.3588
		1.00	.1142	.8358	.5898	.1604	.4199

A_{TOTAL} = Area reqd @ T/O power, S.L. STD. Static

TABLE 3-4 PROPULSION SYSTEM CHARACTERISTICS (Cont'd)

WARM CYCLE - ENGINES 4, 5, & 6 - $V_{TIP} = 640$ FPS

ENGINE NO.	ΔP_{3-5} P_{T3}	THRUST RECOV.	S.L. STD T/O POWER			6000 ft., 95° T/O POWER	
			A_{TOTAL} ESHP	ESHP W_F	ESHP F_{Nrated}	A_{TOTAL} ESHP	ESHP I_{rated}
			IN^2/HP	HP/LB/HR	HP/LB	IN^2/HP	HP/LB
4	.5	.80	.1366	.7289	.4800	.1917	.3419
		.90	.1138	.8735	.5760	.1594	.4113
		1.00	.0976	1.019	.6720	.1363	.4808
	.15	.80	.1661	.6642	.4380	.2340	.3109
		.90	.1376	.8019	.5288	.1932	.3765
		1.00	.1174	.9396	.6196	.1646	.4420
	.25	.80	.2086	.5905	.3894	.2955	.2729
		.90	.1713	.7189	.4741	.2417	.3360
		1.00	.1454	.8474	.5588	.2045	.3971
5	.5	.80	.1643	.7346	.4543	.2272	.3287
		.90	.1357	.8899	.5504	.1872	.3989
		1.00	.1155	1.045	.6464	.1591	.4691
	.15	.80	.2034	.6584	.4072	.2817	.2941
		.90	.1666	.8042	.4974	.2301	.3600
		1.00	.1410	.9500	.5875	.1945	.4259
	.25	.80	.2626	.5695	.3522	.3645	.2537
		.90	.2124	.7041	.4355	.2940	.3146
		1.00	.1783	.8388	.5187	.2463	.3755
6	.5	.80	.2293	.7256	.4092	.3108	.3018
		.90	.1855	.8971	.5058	.2511	.3737
		1.00	.1557	1.069	.6025	.2106	.4456
	.15	.80	.2970	.6216	.3505	.4022	.2588
		.90	.2367	.7601	.4398	.3200	.3253
		1.00	.1967	.9385	.5292	.2657	.3918
	.25	.80	.4140	.4977	.2806	.5596	.2076
		.90	.3216	.6406	.3612	.4340	.2677
		1.00	.2629	.7836	.4418	.3544	.3278

$$K_t = \text{total duct loss coefficient} = \Delta P_t / q$$

$$= (K_b + K_f)$$

K_b = bend loss coefficient, assigned to each bend and fitting individually, based on various references

$$K_f = 4C_f(L/D_h)$$

L = duct length

D_h = hydraulic diameter = $4A/P$

A = cross-sectional area

P = wetted perimeter

C_f = duct friction coefficient, determined from reference (2) as listed on the following page.

The pressure drop calculation is based on the system layout, Figure 3-16. Values of K_b for this system were assigned as follows:

<u>Item</u>	<u>K_b</u>	<u>Ref</u>
(1) Engine expansion bellows	0.02	(1)
(2) Diverter valve	0.10	(2)
(3) Compound bend, engine to rotor mast		
a. 45° in horizontal plane, $R/D = 2$, $Re = 1,084,178$ (based on engine duct diameter)	0.08	(2)
b. 93° in vertical plane, $R/D = 1.7$, $Re = 1,532,304$ (based on mast diameter)	0.16	(2)

(4)	Rotating seal	0.05	(1)
(5)	90° bend (rotor head) $R/D = 0.7$	0.30	(2)
(6)	Spherical joint	0.05	(1)
(7)	Transition to blade ducting	0.10	(1)
(8)	Expansion bellows, 10 at 0.024	0.24	(1)
(9)	Cascade	<u>0.15</u>	(3)
		1.25	

Different reference material gives slightly different values; however, it is estimated that a realistic K_b will be within 20 percent ($\pm .15$) of the nominal value, giving a probable K_b range of 1.10 to 1.40. A value of 0.003 was similarly determined for C_f , subscript. Assuming a variation of ± 20 percent on the C_f number gives a probable range of 0.0024 to 0.0036.

The system can be scaled up or down without affecting K_b ; however, K_f will be strongly influenced by physical geometry. The geometry of the blade is, in turn, influenced by the thrust recovery and pressure drop of the design. Therefore, matching a physical geometry with a pressure drop is an iterative process, but a rapidly converging one.

Figure 3-17 shows the pressure drop calculated for the point design, hot cycle, high speed helicopter, based on 90 percent thrust recovery

-
1. VAD in-house data.
 2. NACA ARR L208 Design of Power Plant Installations, John R. Henry, 1944.
 3. McCormick, Barnes W. Aerodynamics of V/STOL Flight, Academic Press, 1967.

HIGH SPEED HELICOPTER

HOT CYCLE ~ ENGINE NO. 1

THRUST RECOV. = 90%

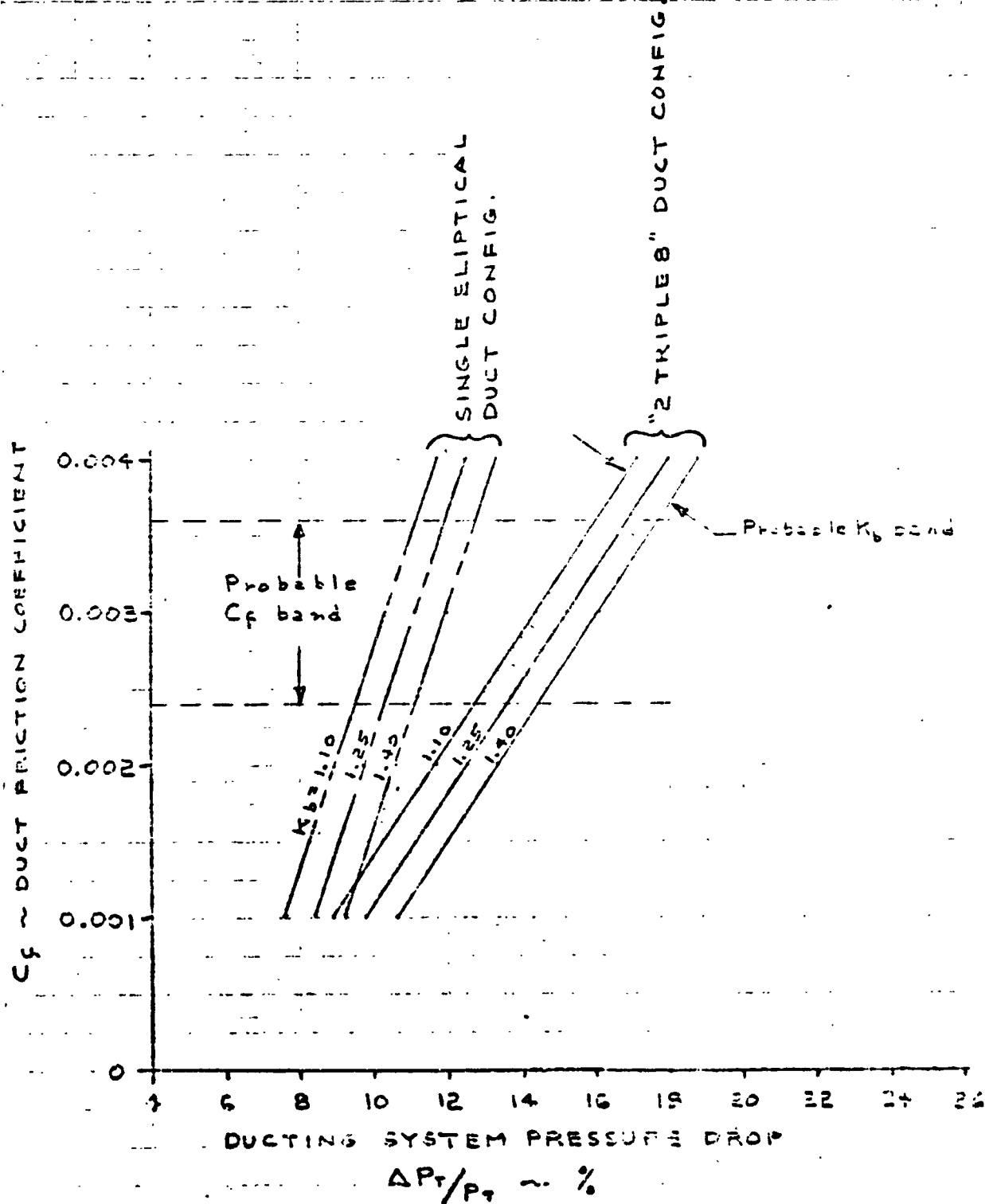
 $M_{duct} = 0.30$ DUCT SIZE BASED ON $\Delta P_T/P_T = 15\%$ 

Figure 3-17

and 15 percent ΔP (configuration No. 2, Table 4-2). In this case, the assumed ΔP of 15 percent is well within the area bounded by the probable limits of C_f and K_b . Also shown is the improvement which could be realized if a single elliptical duct were used instead of the "2 triple 3" configuration shown on Figure 3-16. This improvement is due to the larger hydraulic diameter of the elliptical duct. (However, there are structural problems associated with the elliptical duct in the hot cycle system.)

Figure 3-18 shows the effect of the ΔP assumed for sizing on the ΔP calculated for the resultant system. The assumption of a low pressure drop results in a small duct area requirement, which in turn permits a small rotor chord. This reduction in chord increases the calculated pressure drop. Considering the 5 percent ΔP case (Table 4-2, Configuration 1), a nominal ΔP of 16 percent is calculated, from which it may be concluded that a ΔP of 5 percent is not likely to be achieved. The 25 percent ΔP case (Table 4-2, Configuration 3) results in larger ducts, with consequent reduction in calculated ΔP to a nominal value of 14 percent. It is evident that a wide range of assumed ΔP 's results in a narrow band of calculated values.

To achieve a probable duct loss of 5 percent requires a reduction in dynamic pressure of one third or a duct Mach number of approximately 0.17 lieu of the 0.3 duct Mach number used throughout this study. Severe penalties in duct weight and blade weight would result from the large areas required to maintain such a low internal velocity.

Figure 3-19 shows the results of similar calculations, assuming engine No. 4 (Table 4-2, No. 14), and Figure 3-20 shows the Engine No. 6 configuration (Table 4-2, No. 12). The larger ducts required with

HIGH SPEED HELICOPTER
HOT CYCLE ~ ENGINE NO.1
THRUST RECOV. = 90%
 $M_{duct} = 0.30$

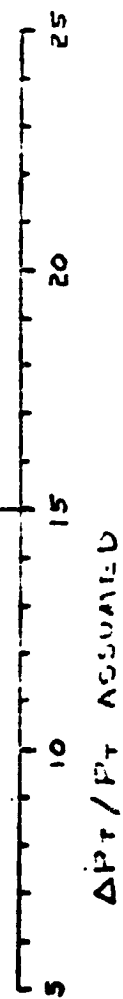
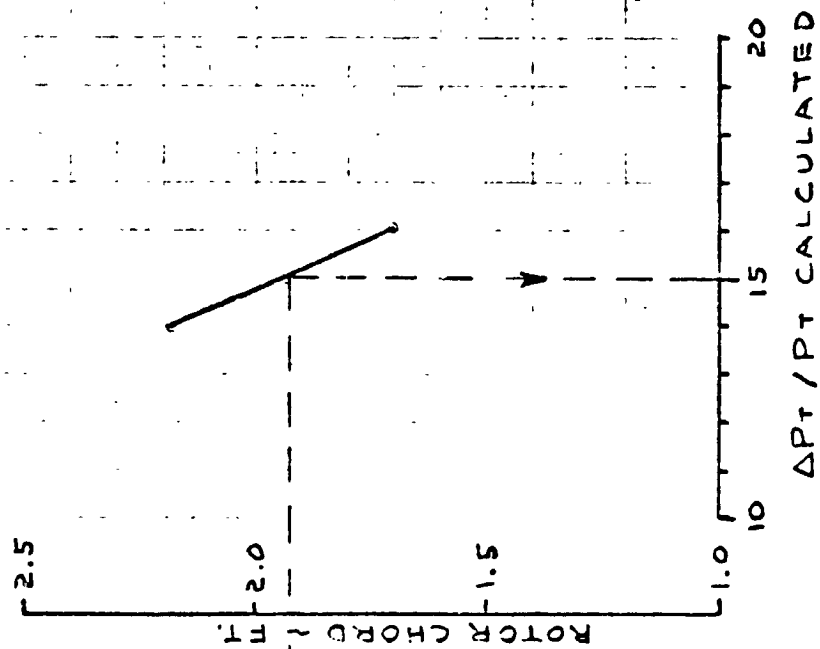


Figure 3-18

HIGH SPEED HELICOPTER
WARM CYCLE - ENGINE NO. 4
THRUST RECOV. = 90%
 $M_{duct} = 0.30$

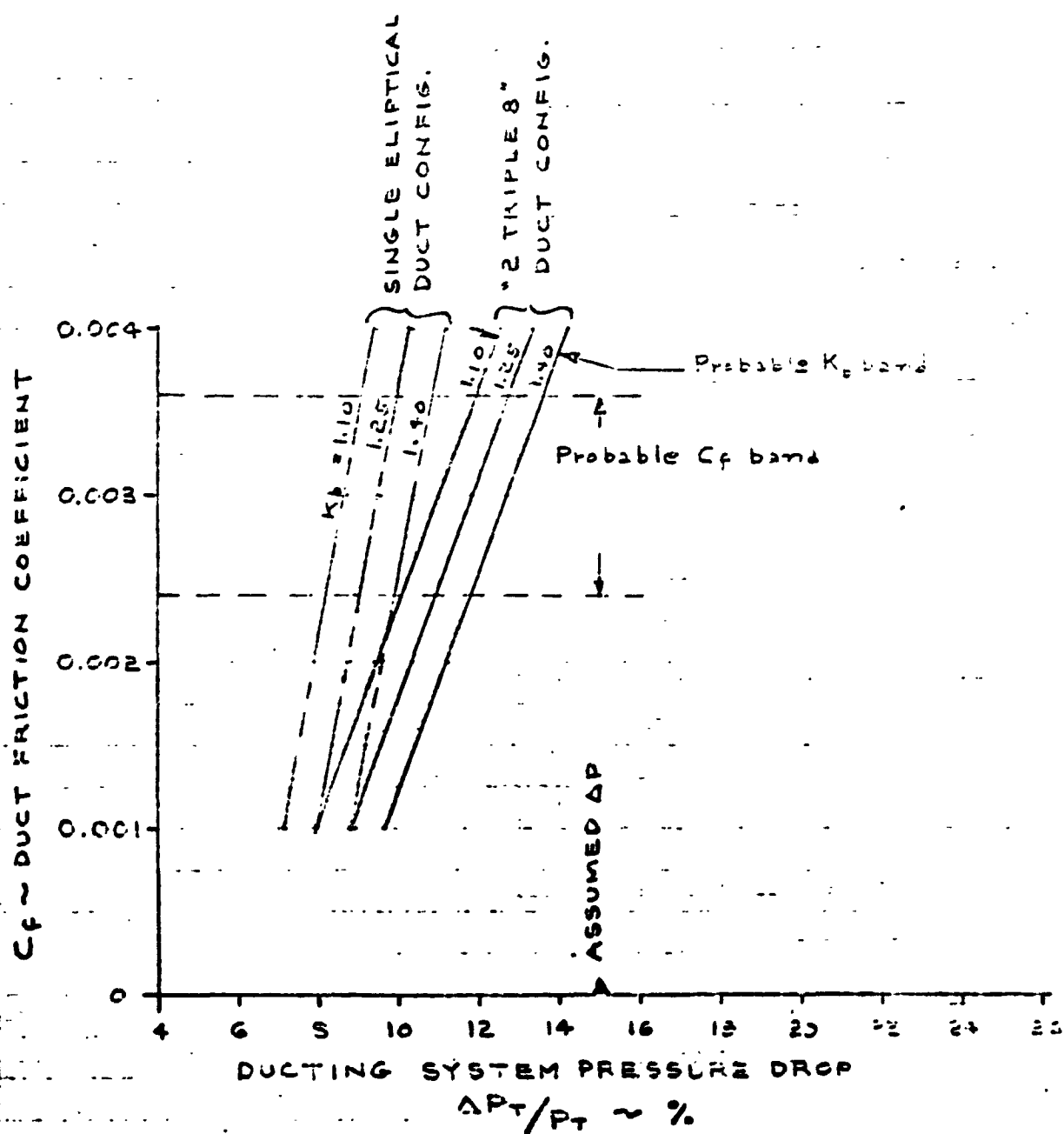


Figure 3-19

3-44

these engines increases the hydraulic diameter and reduces the pressure drop. These systems were not iterated to a match; however, considering the insensitivity effect previously noted, it is reasonable to assume that the Engine No. 4 system would have a ΔP of about 12 percent, and the Engine No. 6 system about 10 to 11 percent.

Figure 3-15, which shows the weight advantage of the hot cycle system, is based on a constant pressure drop, whereas the above discussion indicates a lower probable drop for the warm cycle system. The ΔP sensitivity study, discussed in Section 3.2 of this report, will show that even if the warm cycle systems are given the weight advantage of a lower ΔP , they are still heavier than the hot cycle systems.

Figures 3-19 and 3-20 also show the improvement due to use of a single elliptical duct. Although the elliptical duct could be more easily adapted to the lower pressure warm cycle systems, its advantage is small. The reason is that the warm cycle ducts are so large that the L/D_h ratio is fairly low in either configuration.

3.1.6 Thrust Recovery Investigation

Thrust recovery, as noted in paragraph 3.1.3, reduces to:

$$TR = V_j \text{ effective} / V_{j \text{ ideal}}$$

Three effects are included in the thrust recovery factor, as follows:

- (a) Nozzle Velocity Coefficient - Nozzle and expansion losses will cause the actual jet velocity to be less than ideal, thus reducing the thrust of the jet.
- (b) Deflection - The thrust vector is deflected by the jet-flap, so that only a component of the thrust is realized in the horizontal plane for rotor torque.

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1

HIGH SPEED HELICOPTER
WARM CYCLE - ENGINE NO. 6
THRUST RECOV. = 90%
 $M_{duct} = 0.30$

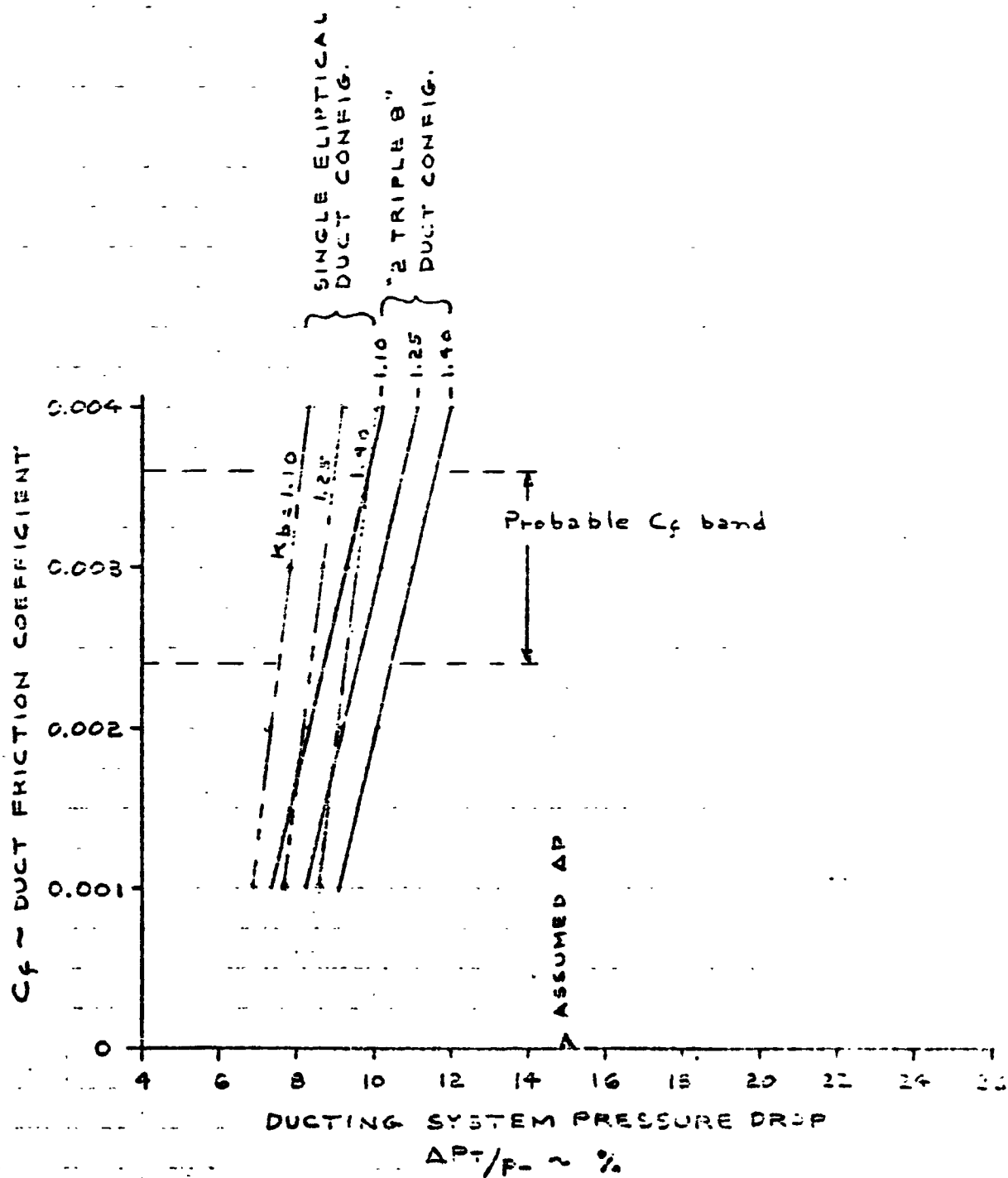


Figure 3-20

- (c) Super-circulation - The jet induces a flow field around the airfoil, which has a thrust component. Therefore, some of the thrust lost by deflection is regained.

Thus far, the thrust recovery factor has defied analysis, and reliance must be placed on the sparse test data available.

Figure 3-21 summarizes some of the theoretical and test data examined in this study.

Line 1 shows the cosine of the total deflection angle, $\alpha + \delta$. This would be the "thrust recovery" if the nozzle velocity coefficient (ϕ Noz) were 100 percent, and there were no supercirculation effect. Williams, Butler and Wood¹ experienced a ϕ Noz of 84 percent. Applying this coefficient to the cosine line gives Line 2.

McCormick² has proposed the following equation, which is a curve fit of the test data of Williams, Butler and Wood:

$$TR = .84 \left[1 - \left(\frac{\alpha + \delta}{100} \right)^3 \right] \text{ with } \alpha \text{ and } \delta \text{ in degrees.}$$

Line 3 is a plot of McCormick's equation. The difference between lines 2 and 3 is the super circulatory thrust increment.

A slight modification of the equation of Evans and McCloud³ gives:

$$TR = \phi \text{ Noz} \left[Sd + (1 - Sd) \cos (\alpha + \delta) \right]$$

In this equation, Sd is a "supercirculation thrust parameter", which can take on values from zero to 1.0. When SD = 0, there is no supercirculation increment, and only the horizontal component of the thrust

-
1. Williams, Butler, and Wood, The Aerodynamics of Jet Flaps, ARC R&M 3304, 1963.
 2. McCormick, Barnes W. Aerodynamics of V/STOL Flight, Academic Press, 1967.
 3. NASA TRD-3028, Analytical Investigation of a Helicopter Rotor Driven and Controlled by a Jet Flap, W. T. Evans and J. L. McCloud III.

THRUST RECOVERY COMPARISON

1

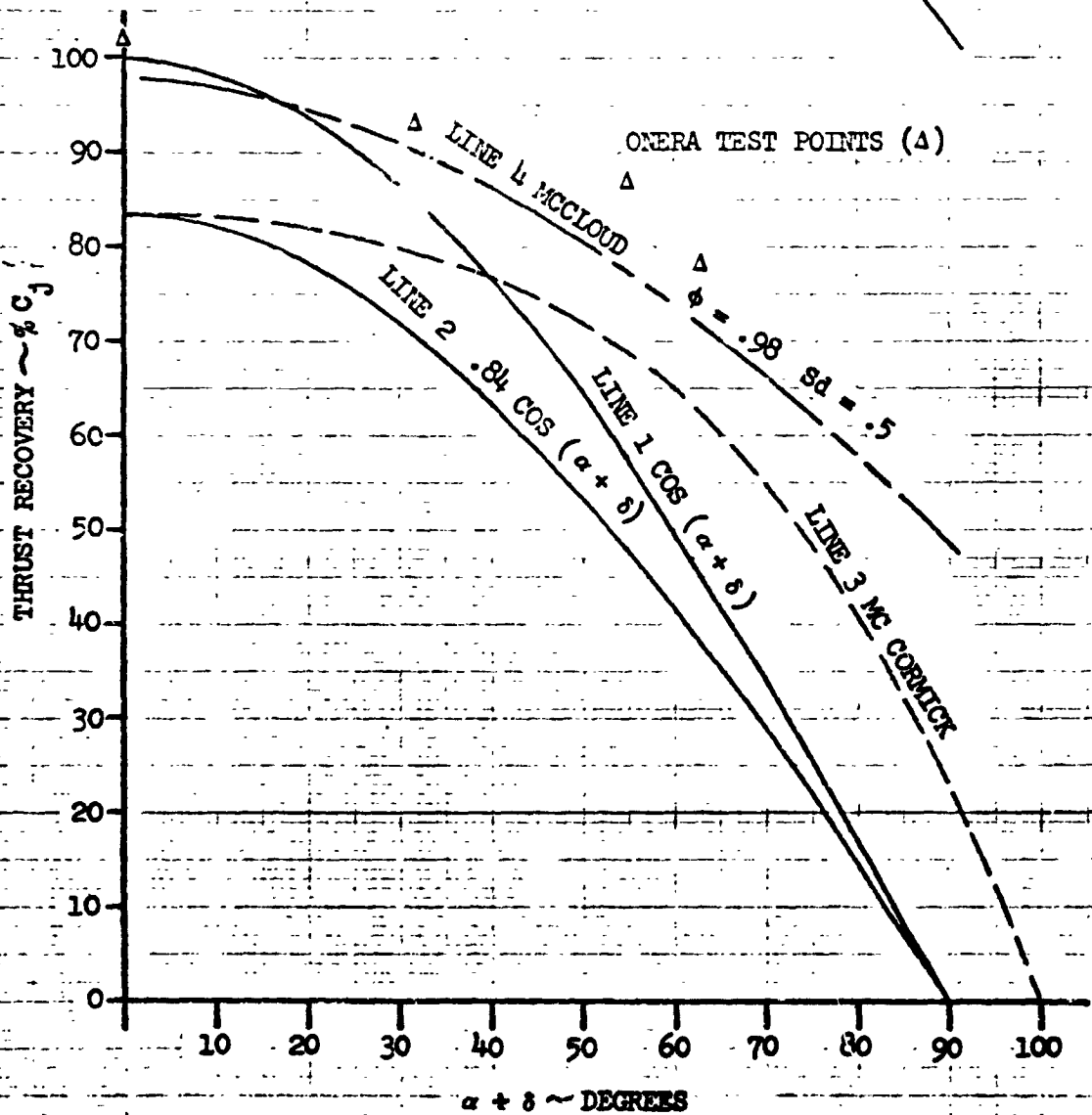
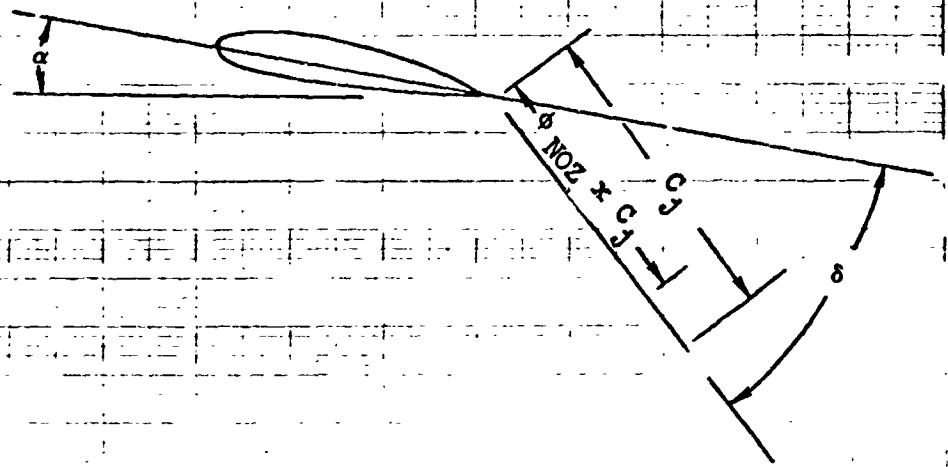


Figure 3-21

vector is realized. When $SD = 1.0$, supercirculation compensates completely for deflection losses, in which case $TR = \phi \text{ Noz.}$

Line 4 is a plot of the preceding equation, with $\phi \text{ Noz} = .98$ and $Sd = .50$. ONERA test points, shown for reference, tend to bear out these values (however, line 3 is also based on test data.)

Figure 3-22 shows the individual effects of deflection, supercirculation, and ϕ nozzle on thrust recovery.

Although none of these curves permit an accurate thrust recovery prediction, it seems reasonable to assume that the 80 percent to 100 percent range considered in this study will probably include all realistic values.

3.1.7 Energy Balance in Jet-Flap System

Figure 3-23 is a bar chart showing the energy balance in a jet-flap propulsion system. The specific system shown is the hot cycle, high speed, point design vehicle (Table 4-2, No. 2).

The first bar shows a rated engine GHP of 6455 HP. Inlet losses, estimated as 1 percent, result of 6390 GHP delivered to the system. Yaw control, leakage, and the pressure drop in the non-rotating ducting result in 5796 GHP at the rotor hub. Centrifugal compression almost compensates for ducting losses in the rotor blade. An additional energy is added to the gas in accelerating it from the hub to the nozzle location, the so called Coriolis acceleration, giving 5986 "total" HP at the nozzle. The system now incurs its greatest loss:

$$\begin{aligned} \text{Jet Efflux Loss} &= \frac{1}{2} M_J (V_J - \bar{V}_T) \\ &= \frac{26.4 \text{ lb/sec} (2780 - 544)^2 \text{ ft}^2/\text{sec}^2}{2 \times 32.2 \text{ ft/sec}^2 \times 550 \text{ ft-lb/sec-HP}} \\ &= 3730 \text{ HP} \end{aligned}$$

THRUST RECOVERY COMPONENT CURVE

EXAMPLE

$$\alpha + \delta = 30^\circ$$

$$S_d = .50$$

$$\phi_{noz} = 95\%$$

$$\text{THRUST RECOV.} = 89\%$$

WHERE:

α = blade incidence, deg.

δ = flap deflection, deg

S_d = supercirculation, thrust param.

ϕ_{noz} = nozzle velocity coef.

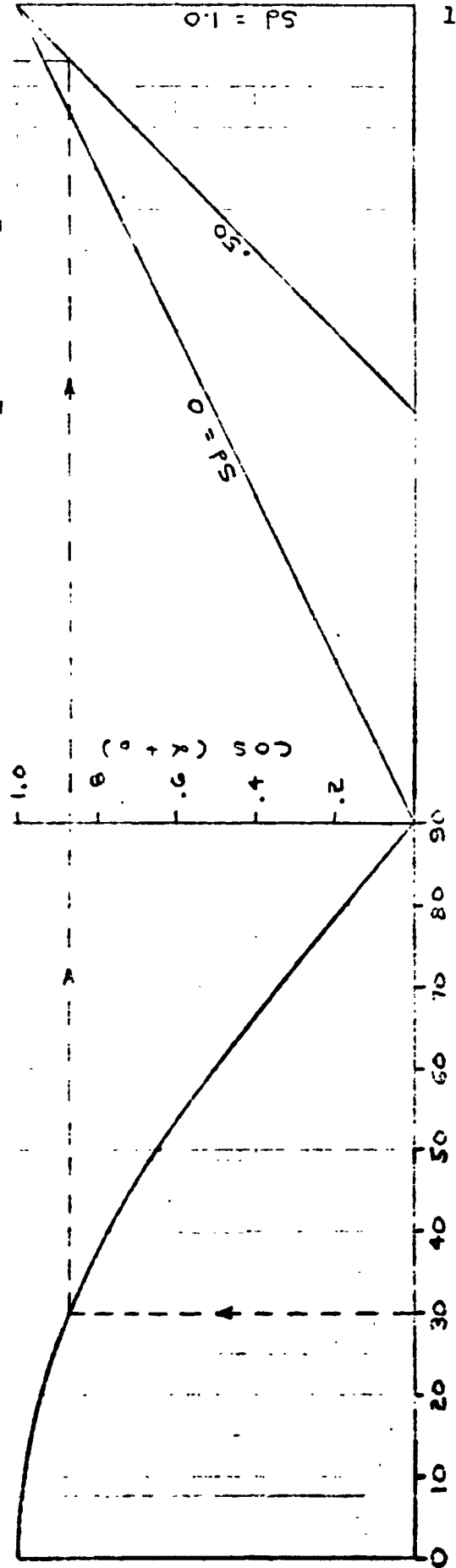
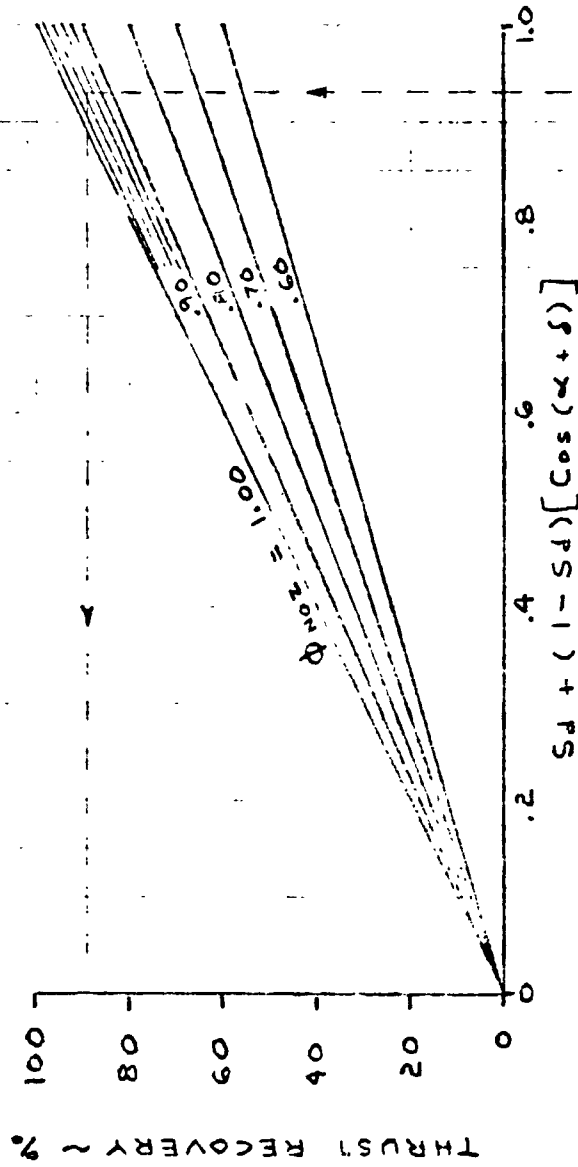
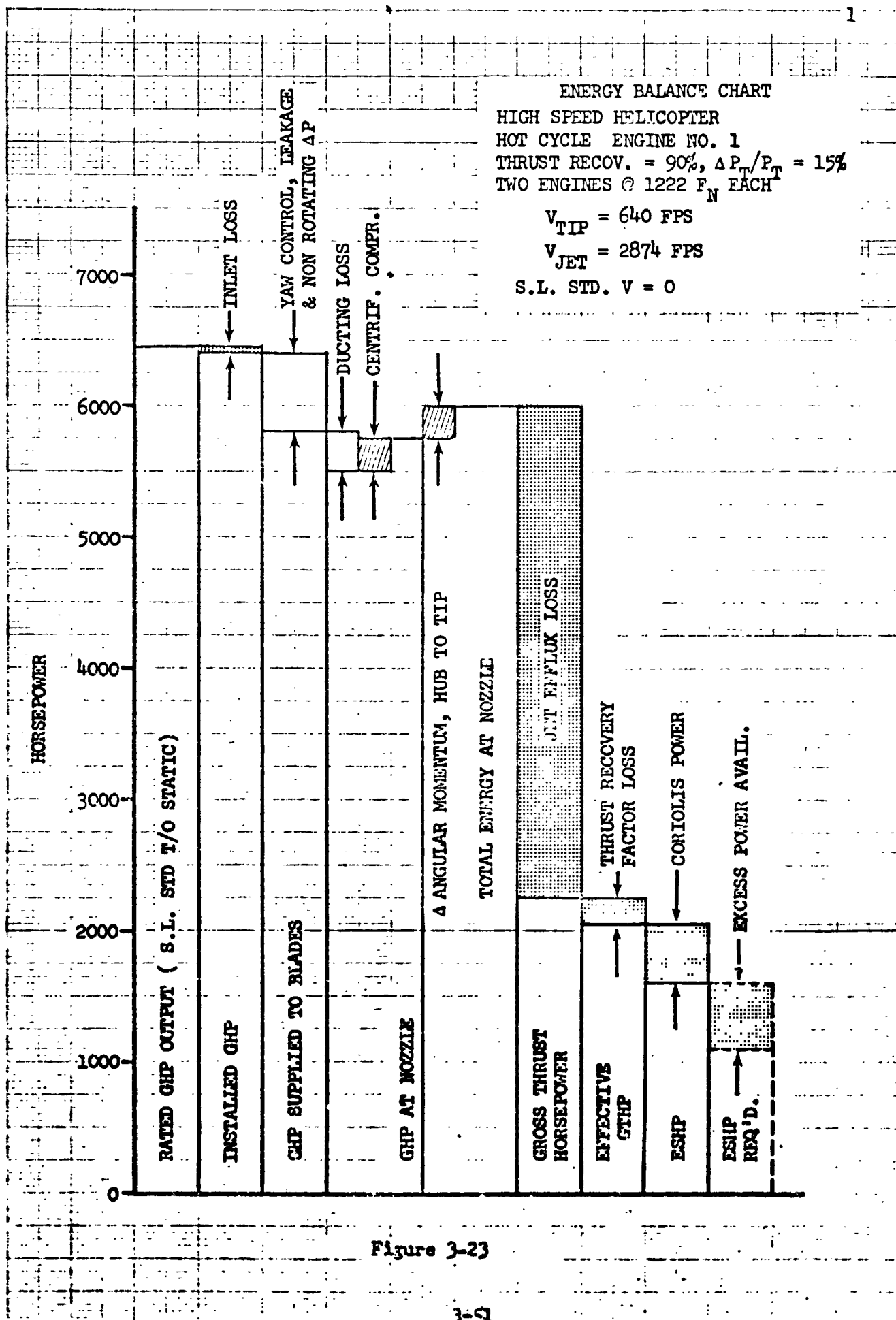


Figure 3-22



This leaves the Ideal Gross Thrust Horsepower of the jet, 2256 HP. For 90 percent thrust recovery, the effective jet gross thrust horsepower = $.90 \times 2256 = 2030$ HP. Some of this power must be returned to the blade for centrifugal compression and Coriolis acceleration, in the amount of Mv_T^2 , in this case 452 HP, resulting in a net ESHP made good of 1578 HP.

Also shown is the sea level hover power required, 1100 HP, resulting in 500 excess horsepower available. This excess power at sea level, standard, results from sizing the engines for hover at 6000 feet on a 95°F day.

3.2 AERODYNAMICS

3.2.1 Rotor Power Requirements

The calculation of rotor power requirements in both hover and cruise is fundamental to the performance prediction methods used in this study as detailed in Appendix A of Volume II. For both the shaft-driven and jet-flapped helicopters, the following equation is used:

$$ESHP = (P_{\text{induced}} + P_{\text{profile}} + P_{\text{parasite}}) \frac{1}{550}$$

$$P_{\text{induced}} = 1.15T_v$$

$$P_{\text{profile}} = \pi R^2 V_T^3 \frac{\phi \bar{C}_d}{8} (1 + 3 \mu^2)$$

$$P_{\text{parasite}} = 1/2 \rho V^3 f.$$

ESHP = Equivalent shaft horsepower.

T = Rotor thrust, lb

v = Induced velocity at rotor disc, ft/sec

ρ = Density, slugs/ft³

R = Rotor radius, ft

V_T = Tipspeed, ft/sec

σ = Solidity

$\bar{C}_d$ = Average profile drag coefficient of rotor blades

$$\bar{C}_d = C_{d_o} + Y\bar{C}_L^2$$

C_{d_o} and Y are functions of the airfoil section. For the jet-flap $Y=0$.

$$\mu = V/V_T$$

V = Free stream velocity, ft/sec

f = Equivalent flat plate drag, ft²

For the shaft-driven helicopter, the tail rotor power requirement in hover is 7% of the main horsepower and is included in the powerplant installation losses. As noted in Appendix A of Volume II, power requirements in high speed flight are corrected for the effect of compressibility on the advancing blade and blade stall on the retreating blade.

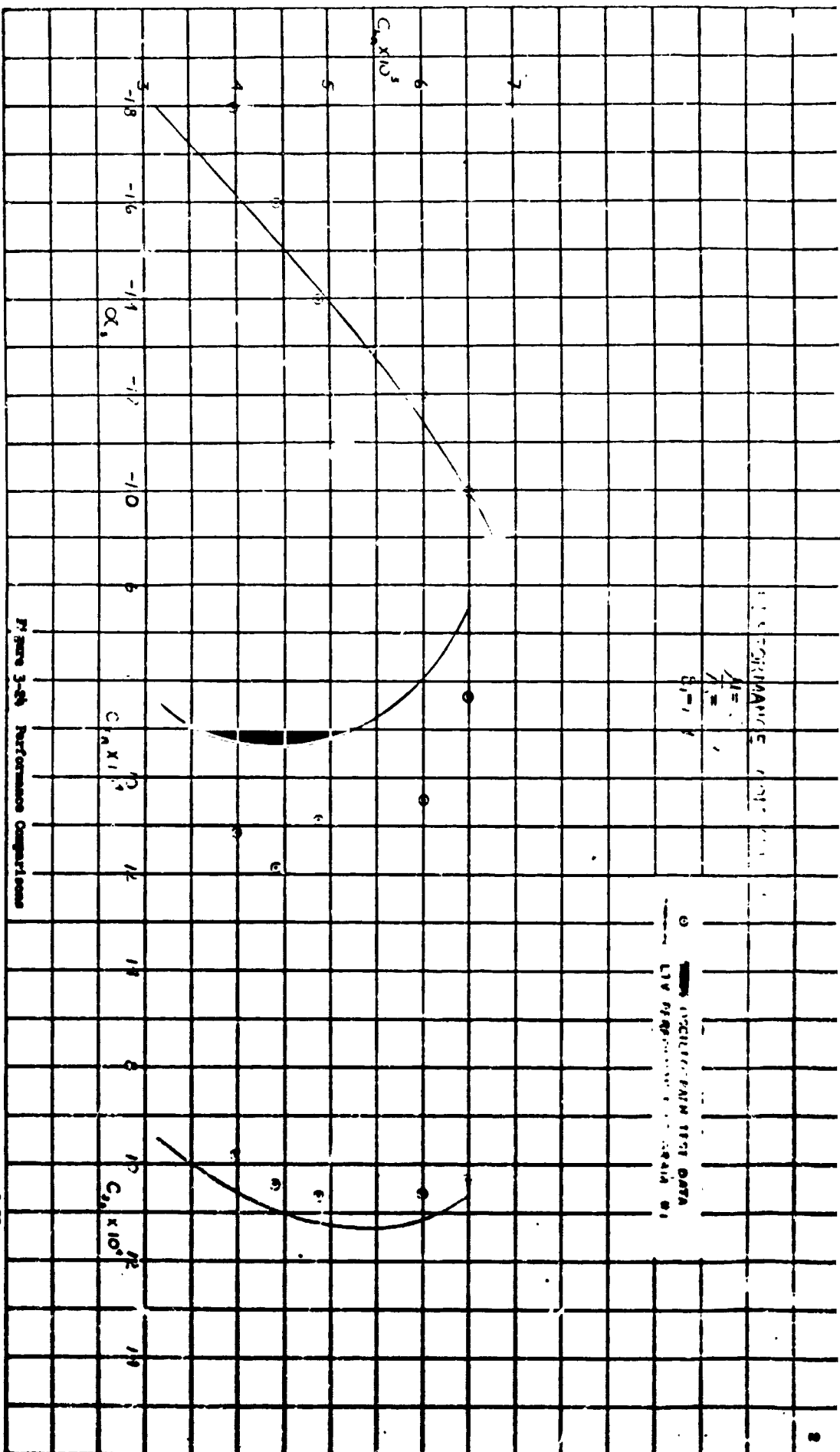
Throughout the parametric portion of the study of the jet-flap helicopter, the bulk of the calculations were accomplished using LTVAC Rotor Performance Program No. 3 and the Mission Performance Program. From the practical standpoint of computer time and complexity, both these programs neglect rotor flapping of trim. Thus rotor performance is governed only by power requirements. The approach is considered valid based on the rotor azimuth survey of local blade conditions and comparisons with CR-114. This methodology of computing power requirements is also employed in LTVAC Rotor Performance Program No. 1. Comparison of data from the First Series Ames Wind Tunnel Tests with Program No. 1, shown in Figures 3-24 and 3-25, further substantiate the use of this method in predicting jet-flap rotor performance.

3.2.2 Drag Analysis

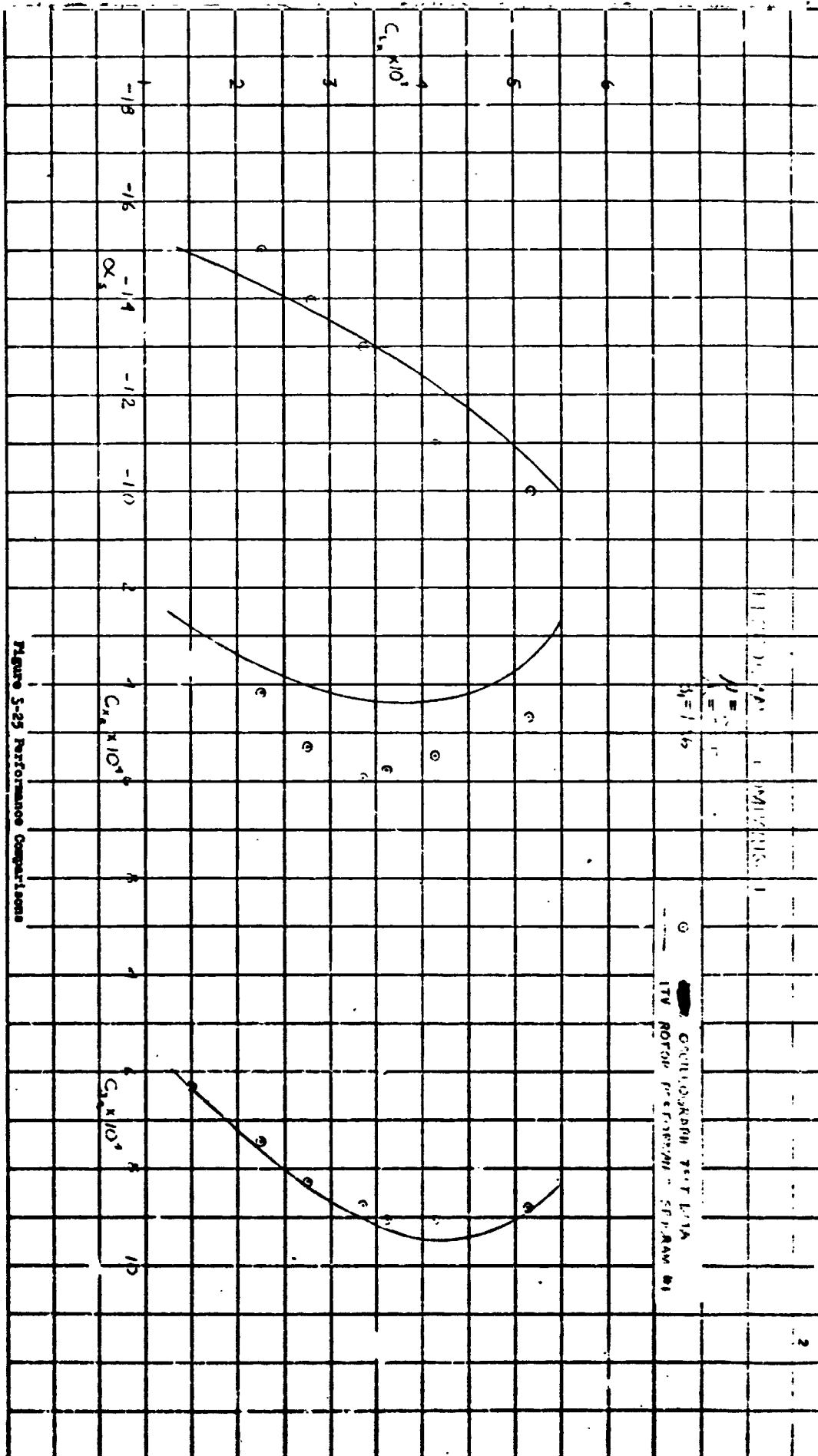
3.2.2.1 Vertical Drag

The following factors are employed in increasing the thrust requirement in hover due to the rotor downwash impingement on the fuselage.

<u>Configuration</u>	<u>Download Factor</u>
High-Speed Shaft	1.05
High-Speed Jet-Flap	1.05
Heavy-Lift Shaft	1.10
Heavy-Lift Jet-Flap	1.04



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These factors are obtained using the dynamic pressure distribution in a rotor downwash from reference (1) and an average vertical drag coefficient from reference (2) based on the affected area. The larger download factor for the heavy-lift shaft-driven helicopter is due to its having the higher disc loading.

3.2.2.2 Frontal Drag

(a) High-Speed Helicopter

The forward flight drag of the vehicles, excluding the rotor is based on the respective drawings included in this report. The method consists of estimating the basic parasite drag coefficient for each major component based on its own wetted area. The coefficients are obtained from reference (3) considering the shape of each component. Table 3-5 presents this drag breakdown for shaft-driven and jet-flap helicopters.

These values compare favorably with published drag levels attained from drag reduction programs for the Bell UH-1B High Performance Helicopter and Lockheed XH-51 Helicopter. The greater hub-nacelle-pylon drag for the jet-flap helicopter is primarily due to the duct requirements creating a larger pylon and hub.

(1) NACA TN 4239

(2) WADD TR 61-124

(3) Hoerner, Sigward F., Fluid-Dynamic Drag, Published by Author, Midland Park, New Jersey, 1958

TABLE 2-5 DRAG BREAKDOWN HIGH-SPEED HELICOPTERS

Shaft-Driven Helicopter Component	C_{d_f}	S_{wet} ft ²	f ft ²
Fuselage	.0043	682	2.93
Vertical Tail	.0074	80	.588
Horizontal Tail	.00544	83	.453
Main Rotor Hub, Engine Nacelles, Pylon + Interference	---		5.27
Tail Rotor Hub	---		.18
Total f (Shaft-Driven)			9.421 ft ²
Jet-Flap Helicopter Component	C_{d_f}	S_{wet} ft ²	f ft ²
Fuselage	.00445	688	3.05
Empennage	.00804	92.2	.74
Main Rotor Hub, Engine Nacelles, Pylon + Interference			8.21
Total f (Jet Flap)			12.0 ft ²

(b) Heavy-Lift Helicopter

Due to the low cruise speed (40 knots) of the heavy-lift shaft-driven helicopter, the power required and thus fuel consumption is essentially insensitive to parasite drag. Actually, the parasite power is less than three percent of the total cruise power. It thus was reasonable to base the drag estimate for both heavy-lift helicopters on the analysis of a similar size crane helicopter reported in reference (4). The following values of equivalent flat plate drag were used:

Crane without Payload	$f = 80 \text{ ft}^2$
Crane with Faired Pod Payload	$f = 100 \text{ ft}^2$
Crane with External Sling Payload	$f = 180 \text{ ft}^2$

The additional drag area, $f = 100 \text{ ft}^2$ of the external sling payload used in the mission performance is representative of a large unfaired load. Any attempt to fair such a bulky load for the short mission radius is impractical.

3.3

Missions

An analysis of high-speed and heavy-lift helicopter uses, presented in Paragraph 2.0 of Volume II, resulted in the selection of the following design missions:

Payload	= 4,000 lb
Radius of action	= 150 NMI
Cruise speed	= 200 kt or NRP speed
Power requirement	= $T/W = 1.0$ at 6,000 ft., 95° F

- (4) Parametric Analysis and Preliminary Design of a Shaft-Driven Rotor System for a Heavy-Lift Helicopter, USAAVLABS Technical Report 66-48 Lockheed-California Company, U.S. Army Materiel Laboratories, Fort Eustis, Virginia, August 1966

Hover time:

At origin	= 5 min
At mid-mission	= 10 min
Warm-up allowance	= 5 min at NRP
Reserve	= 10% initial fuel

The 4000 lb. payload selected permits an adequate weapons capability for an escort/weapon delivery mission. The radius of action was originally recommended as 200 NM, but early studies indicated that the payload/speed/radius combination of 4000 lb/200 kt/200 NM resulted in a divergent fuel required/fuel available. The radius was therefore reduced to 150 NM. Hover requirements, warm-up allowance and reserves are those normally used. Power requirements are determined from the necessity to provide vertical takeoff and landing.

HEAVY-LIFT HELICOPTER (HLH)

Payload	= 40,000 lb
Radius of action	= 10 NMI
Cruise speed	= 40 kt
Power requirement	= T/W = 1.1 at SL Std
Warm-up and Hover	= 5 min
Mid-mission hover	= 2 min
Reserve	= 10% initial fuel

This mission includes most cases where it is believed a crane-type vehicle would be used for special operations such as retrieval of downed aircraft, unloading of ships where port facilities are inadequate, movement of heavy equipment across areas impassable by surface means, etc. The payload requirement was at first set at 50 tons but preliminary studies indicated that the power requirements would be too large. Therefore, the original requirements were revised to allow the parametric study to be conducted around a power requirement which is more realistic for the 1975-1980 time period.

3.4 Weight Methodology

Methods which may be used to derive and justify helicopter component weights are technically limited at present by the amount of detail design. Detail design for the subject study was very limited, requiring only enough technical depth to permit the selection of the mission best suited to utilize the capabilities and potentials of the jet-flap rotor. Thus, sizing of the propulsion arrangement of each concept has received the major emphasis. The enclosing structure and fixed equipment items received less detailed attention. While normally, three standard estimating methods - semianalytical, statistical, and component sizing are used, only one, statistical analysis was used extensively in this study to derive the estimated weights. The method of statistical analysis consists of using statistical weight estimating equations and certain design data in deriving the estimated weights. Statistical weight estimating equations are used for as many as possible of the major helicopter components of each concept studied. The statistical weight estimating equations were selected so that the same equation was used for identical components of both concepts, thus eliminating any optimism or conservatism which is induced when identical component weights are derived by two different statistical weight estimating equations.

It was necessary to deviate from this philosophy only once in deriving the estimated weight; this conflict occurred in deriving the weight of the transmission system. For the shaft-driven rotor, a conventional shaft and gearbox arrangement is used. The jet-flap rotor concept, however, is driven by a hot gas system plus shaft and gearbox arrangement.

For those few situations where it was not possible to use statistical methods of weight estimating, component weights were derived by other methods. These methods consist of using data obtained from:

- . Vendors
- . Contemporary helicopters
- . Calculations based on preliminary drawings

Basic weights group input to the parametric study were fuel available values as functions of gross weight. Fuel available data were obtained from a parametric weight analysis routine using the statistical weight estimating equations described in Volume II. Parameters influencing the fuel available were gross weight, diameter, disc loading, and solidity.

3.5 STRUCTURAL AND DESIGN CONSIDERATIONS

3.5.1 Preliminary Design Criteria

The structural design criteria for both the high-speed helicopter (HSH) and the heavy-lift helicopter (HLH) were chosen using MIL-S-8698 (ASG) as a guide. In accordance with the study plan's philosophy, these criteria are presented to define both the shaft-driven and jet-flap rotor helicopters sized for both the HLH and HSH missions. Specific portions which are peculiar to either the shaft-driven or the jet-flap helicopter are so noted. The helicopter structures shall be

capable of supporting without failure the ultimate loads resulting from the loading conditions and ultimate factors safety specified below. Allowable stress values to be used in the stress analysis will be those taken from approved Government publications such as MIL-HDBIC-5, or various NASA or Bureau of Standards Reports.

Factors-of-safety used were:

1. The minimum yield factor will be 1.0.
2. The minimum ultimate factor of safety will be 1.5.

The helicopter and its components will be designed for a minimum fatigue life of 1,000 hours with a desirable service life of 3,600 hours.

The basic mission of the HSH puts the vehicle in Class I helicopter of MIL-S-8698(ASG). Design limit load factors at the cg of the helicopter are +3.5 and -0.5.

The basic mission of the HLH puts the vehicle in Class III of MIL-S-8698(ASG). Design limit load factors at the cg of the helicopter are +2.5 and -0.5.

3.5.2 Design Studies

Preliminary design, structural, materials, and dynamics studies were conducted to assist in determining realistic structural weights of the jet-flap helicopter to support the performance parametric studies and to identify some of the more critical design and technological problem areas. These studies are discussed in Volume II. Because of the many different parametric designs, typical point designs were chosen for the specific studies and to illustrate representative design concepts. General arrangement drawings of the high-speed shaft-driven

helicopter, hot cycle, and cold cycle jet-flap helicopters are illustrated in Figures 3-26, 3-27, and 3-28. Heavy-lift concepts, illustrated in Figures 3-29, 3-30, and 3-31 are the shaft-driven, cold cycle jet-flap and warm cycle jet-flap helicopters.

Discussion of some of the supporting design studies are included Volume II. Included in the structural design studies were hub and mast design, blade design, and blade ducting arrangements. Stress and load analyses, including blade structural sizing and static droop, were conducted on the rotor system. Drive system studies were conducted. Preliminary dynamic assessments in the areas of rotor dynamic behavior, vibration, and acoustics were made.

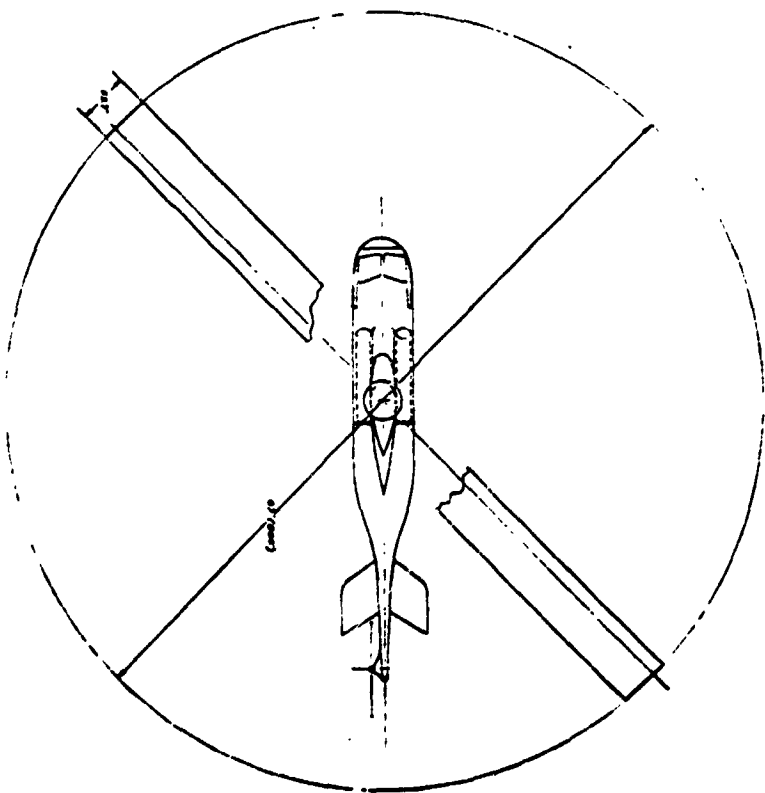
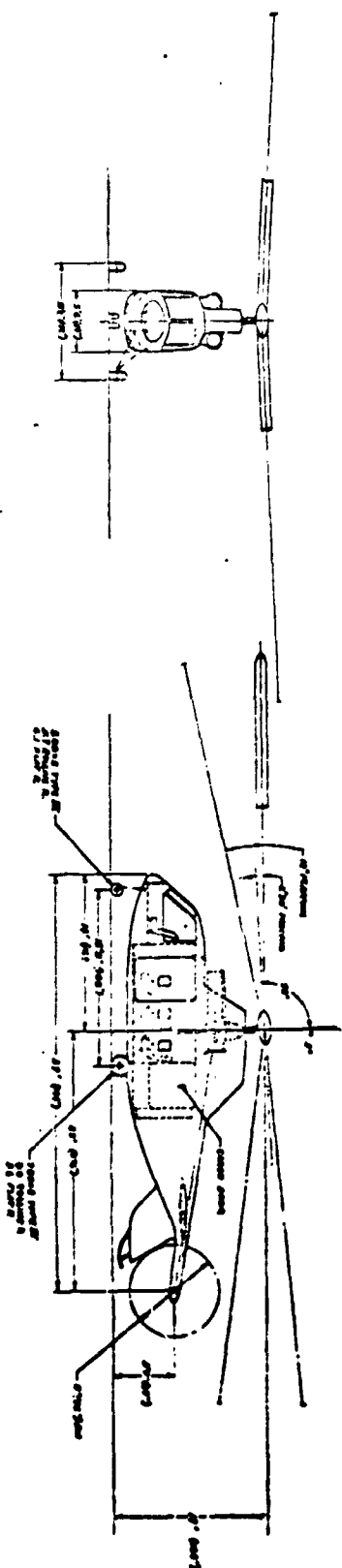
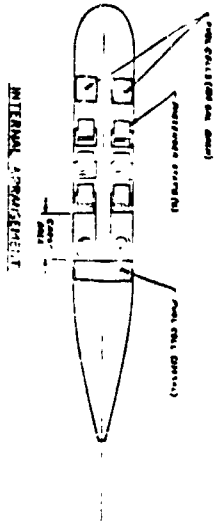
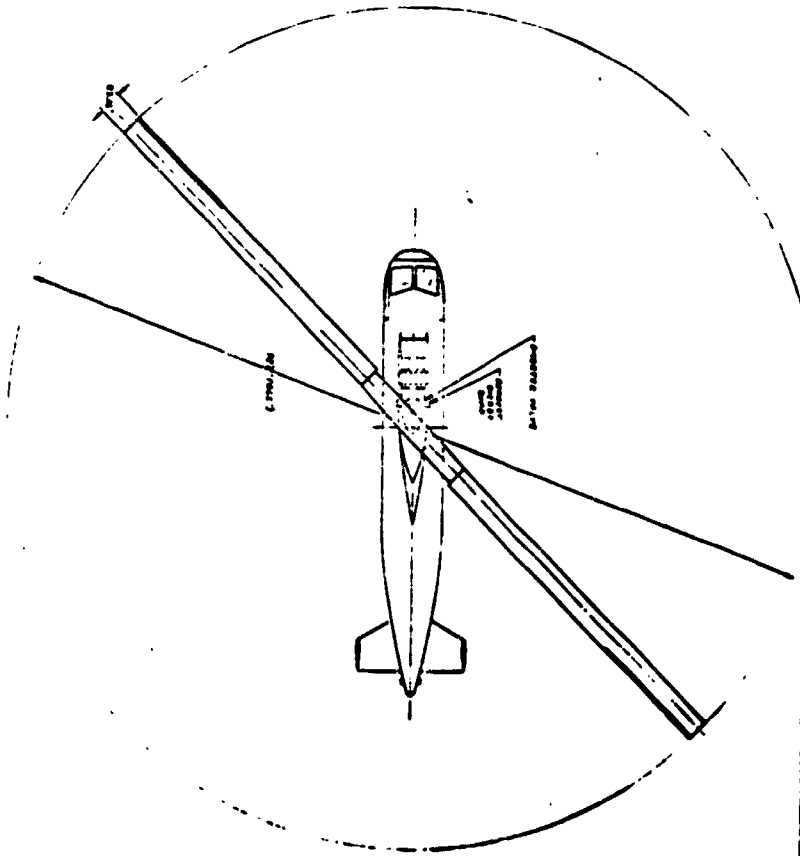


Figure 3-26
3-65/66 -

[illegible]



INTERNAL ARRANGEMENT



NOTES:
1. The ship is a 1000-ton ship.
2. The ship is a 1000-ton ship.
3. The ship is a 1000-ton ship.
4. The ship is a 1000-ton ship.

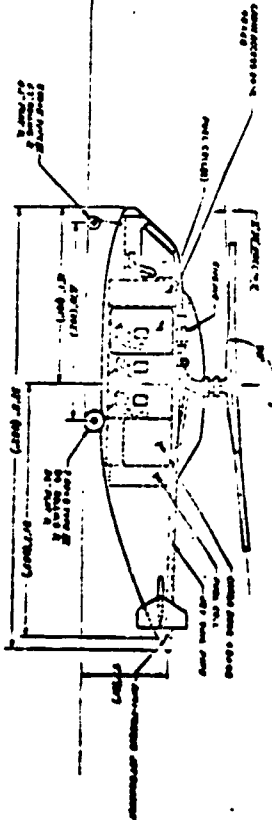
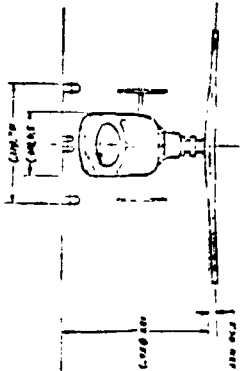


Figure 3-27
3-61/68

1	2	3	4	5	6	7	8	9	10
11	12	13	14	15	16	17	18	19	20
21	22	23	24	25	26	27	28	29	30
31	32	33	34	35	36	37	38	39	40
41	42	43	44	45	46	47	48	49	50
51	52	53	54	55	56	57	58	59	60
61	62	63	64	65	66	67	68	69	70
71	72	73	74	75	76	77	78	79	80
81	82	83	84	85	86	87	88	89	90
91	92	93	94	95	96	97	98	99	100

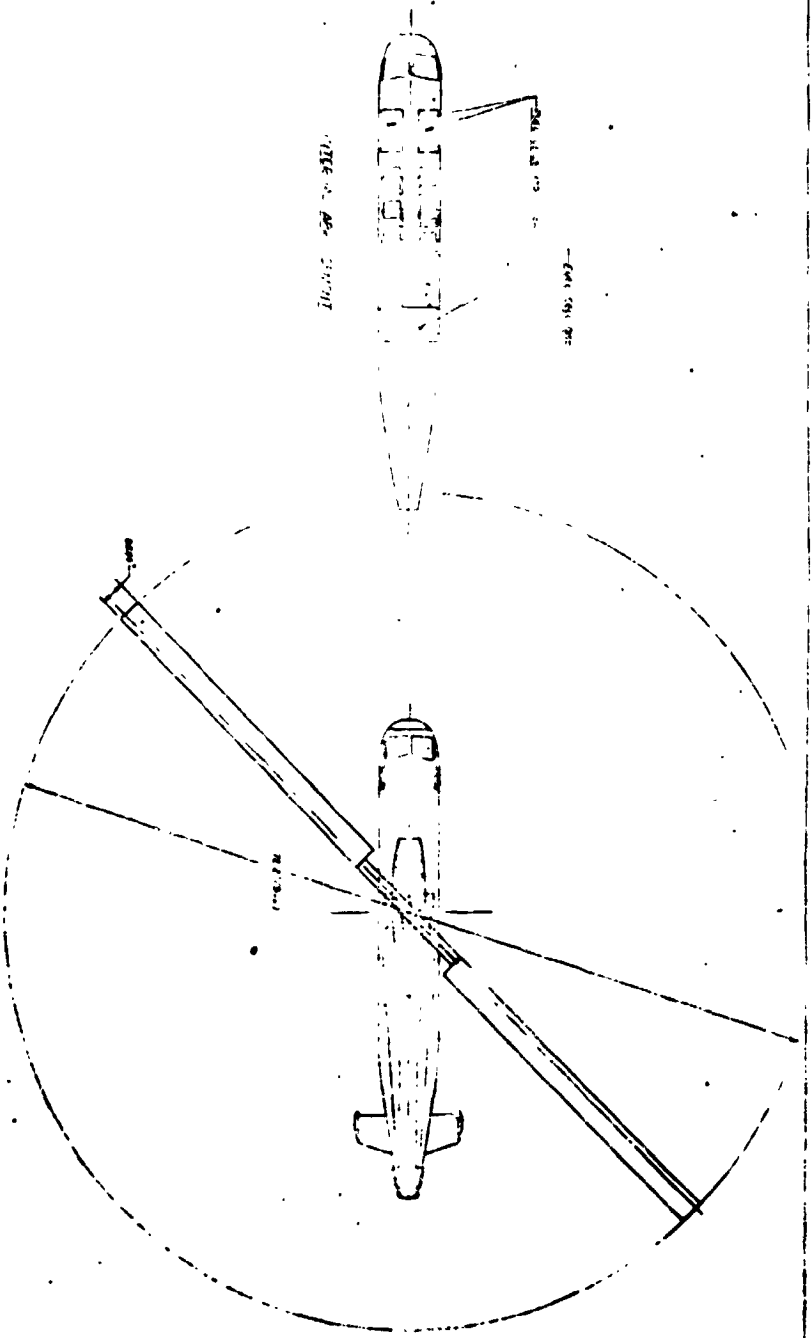
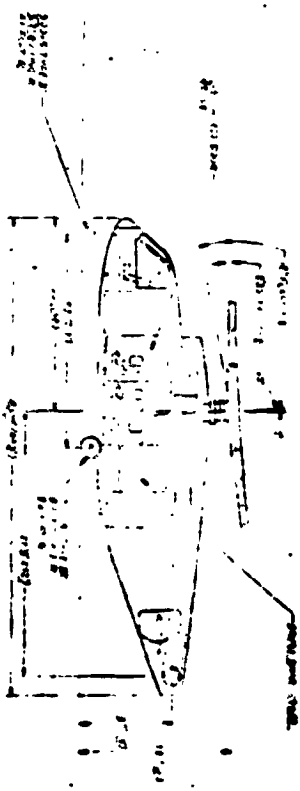


Figure 3-26
3-6970



1	2	3	4	5	6	7	8	9	10
11	12	13	14	15	16	17	18	19	20
21	22	23	24	25	26	27	28	29	30
31	32	33	34	35	36	37	38	39	40
41	42	43	44	45	46	47	48	49	50
51	52	53	54	55	56	57	58	59	60
61	62	63	64	65	66	67	68	69	70
71	72	73	74	75	76	77	78	79	80
81	82	83	84	85	86	87	88	89	90
91	92	93	94	95	96	97	98	99	100

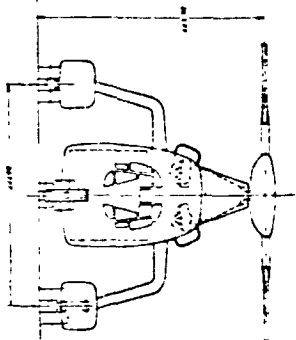
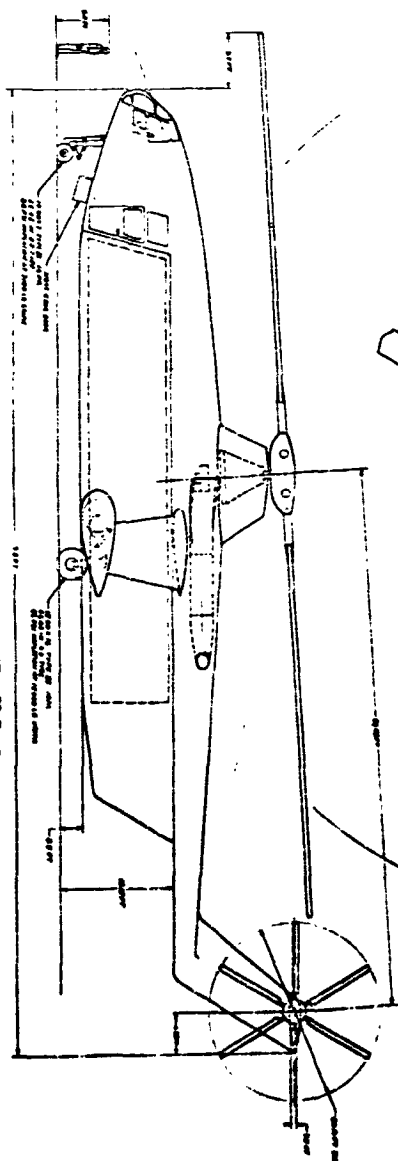
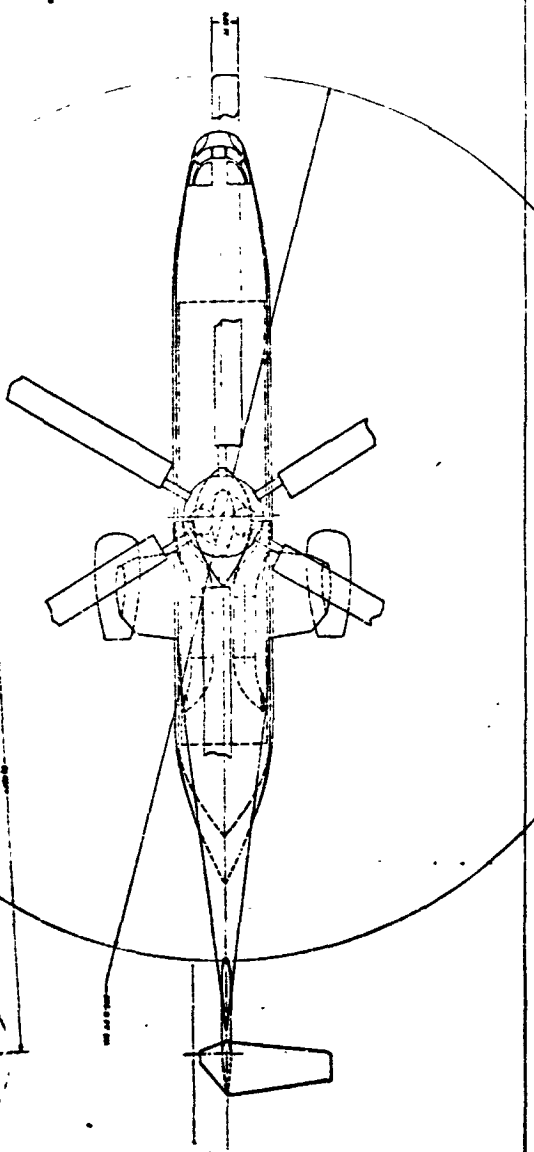


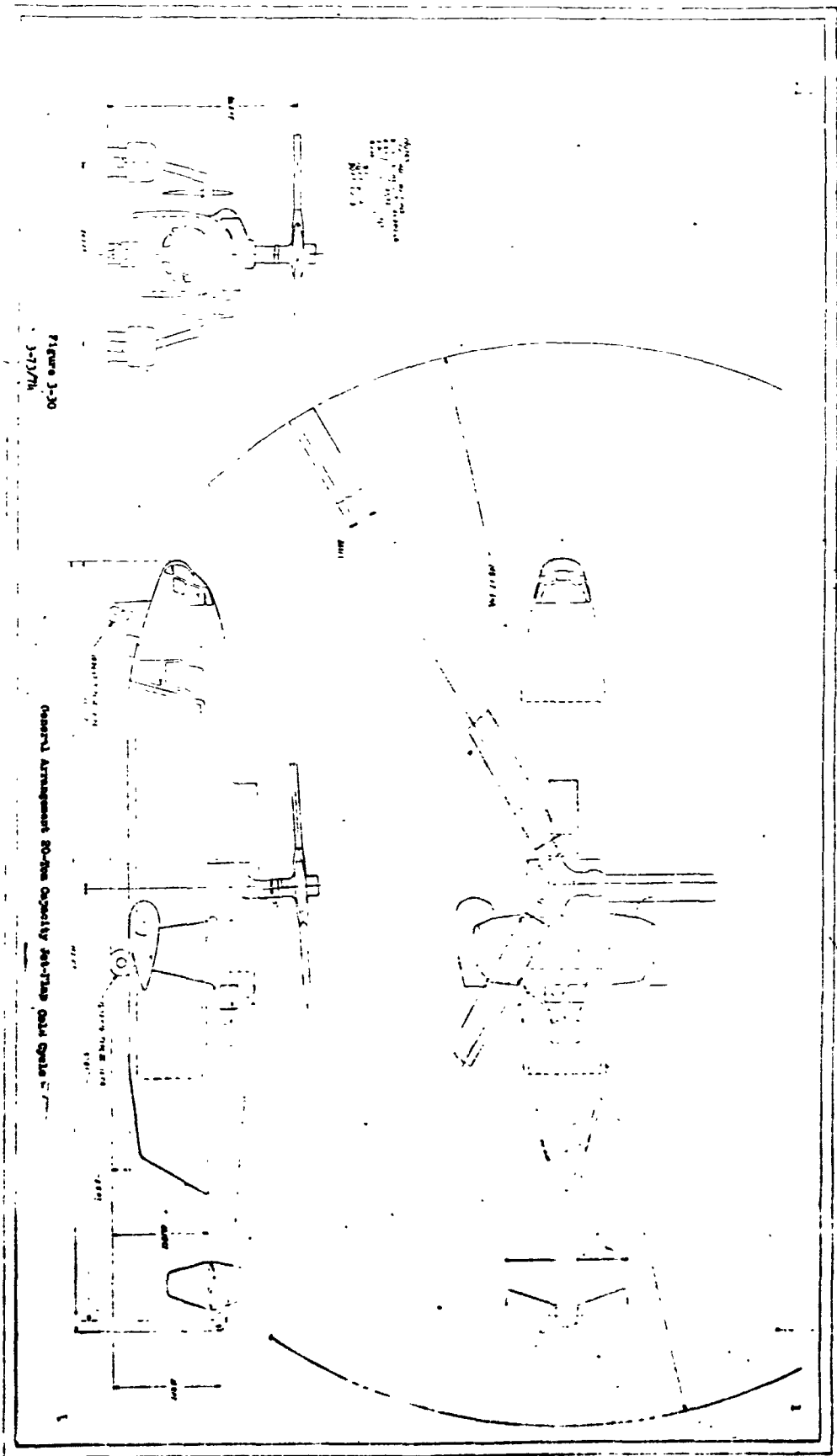
Figure 3-29
3-11/72

NOTE:
1. The ship is shown in
a simplified manner to
show the general
arrangement of the
propeller and rudder
assembly.



General Arrangement, M1 80-000
Capacity 8000 Tons





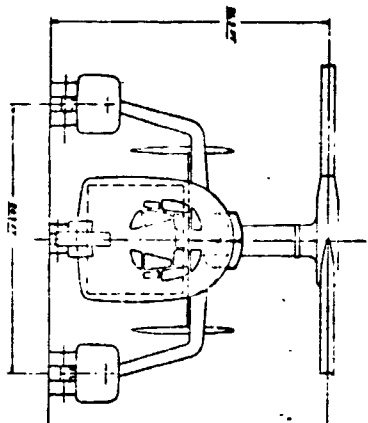
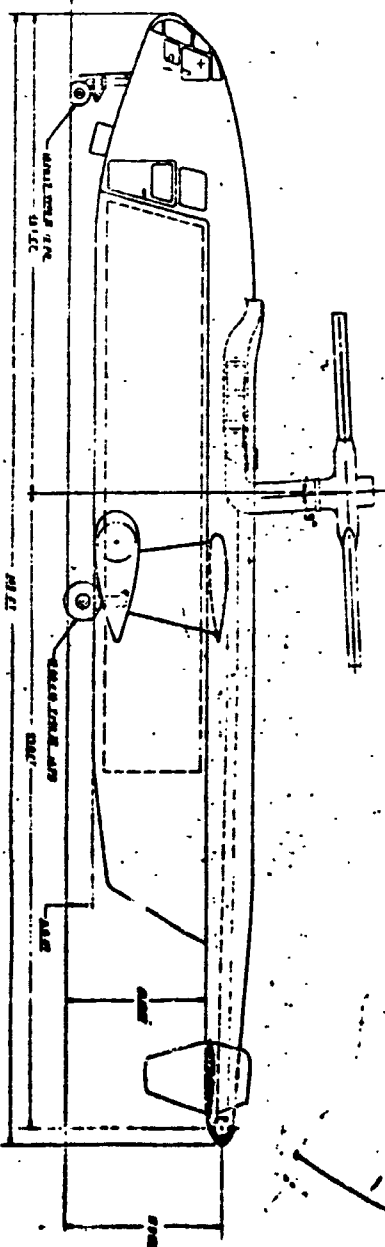


Figure 3-31
3-75/76



General Arrangement 80-Ton Capacity Jet-Flag Main Cycle

Notes:
1. The ship is designed for a maximum speed of 10 knots.
2. The ship is designed for a maximum range of 1000 miles.
3. The ship is designed for a maximum endurance of 10 days.
4. The ship is designed for a maximum crew of 100 personnel.
5. The ship is designed for a maximum payload of 100 tons.

4.0 RESULTS

4.1 PARAMETRIC STUDIES

4.1.1 Philosophy

A parametric analysis has been performed to optimize several helicopters for comparison in this application study. The resulting designs were optimized on minimum gross weight capable of satisfying mission requirements. At each point in the parametric studies, an initial gross weight was assumed in addition to a combination of geometric parameters representing rotor solidity and diameter. The power needed to satisfy the hover and cruise requirements was computed, and the fuel required to perform the design mission was determined. A weight analysis of the vehicle was then conducted to predict fuel available. A plot of this fuel available versus gross weight was compared with a plot of the fuel required versus gross weight, leading to the optimum configuration. This procedure was repeated for various combinations of rotor solidity and diameter to obtain the optimum (minimum) gross weight point design.

Based on a preliminary sizing study, the range of gross weights for each concept was chosen to bracket the optimized gross weights. The range of rotor diameter and solidity was selected to achieve accepted values of average rotor lift coefficient in hover according to the following equation:

$$\bar{C}_L = \frac{3C_T}{\sigma} = \frac{6\pi}{\rho V_T^2 BcR}$$

where T = Rotor Thrust, lb

ρ = Air density, slugs/ft³

V_T = Rotor tip speed, ft/sec

B = No. of Blades

c = Blade chord, ft

R = Rotor radius, ft

4.1.2 Discussion of Parameters

4.1.2.1 High-Speed Shaft-Driven Helicopter

A rotor tip speed, $V_T = 640$ ft/sec, was selected to: (1) achieve a maximum rotor tip Mach number, $M_{TIP} = .95$, at the desired maximum dash speed, $V_{max} = 250$ kt; and (2) maintain sufficient rpm to limit coning angle. The number of blades was set at two. The combination of these parameters yields rotor lift coefficients of reasonable value which bracket a $\bar{C}_L = .5$.

4.1.2.2 High-Speed Jet-Flap Helicopter

Based on recommendations by Dorand personnel during a visit to their facility, the following values were held fixed through the study.

. Limit Tip Mach No. = .95

. No. of Rotor Blades = 2

Since the thermodynamic efficiency of a jet-flap rotor is benefited at high tip speed, a maximum value of $V_{tip} = 640$ ft/sec was used, being consistent with a limit tip Mach No. = .95 at the desired dash speed of 250 knots.

As shown in the propulsion section, a low pressure ratio and, thus, large blade duct cross-sectional area increases ideal thermodynamic

efficiency. Therefore, for a selected solidity, a large chord and, thus, minimum number of blades is desirable according to the following equation:

$$\sigma = \frac{BC}{\pi R}$$

where σ = Blade solidity

B = Number of blades

C = Blade chord, ft

R = Rotor radius, ft

Therefore, in the further interest of simplicity, low weight, and low cost, a two-bladed rotor is desirable. The range of parameters was selected to yield average rotor lift coefficients consistent with results from the Ames First Series Wind Tunnel Tests of the Dorand rotor.

4.1.2.3 Heavy-Lift Shaft-Driven Helicopter

A limited parametric study was employed in defining the heavy-lift shaft-driven helicopter. Based on the Sikorsky S-64 Sky-Crane, a six-bladed rotor with a disc loading of 10.3 psf and a solidity of 0.106 was employed as typical of present-day heavy-lift helicopters. Thus, by fixing the disc loading, a specific diameter is matched to each of the gross weights used in the study.

It should be noted that for the gross weights corresponding to the 20-ton payload vehicle, the 10.3 psf disc loading results in a rotor diameter of around 100 ft. It will later be shown that the rotor of the optimum heavy-lift jet-flap vehicle is 145 ft with a lower disc loading. Because of torque, reduction gear ratio, and tail rotor-main rotor interference considerations, it is believed that rotor diameters around 100 ft represent

a reasonable maximum for shaft-driven vehicles of this class. A tipspeed of 700 fps was selected in order to yield an average rotor lift coefficient, $\bar{C}_L = 0.5$, consistent with present technology. By using this approach for the shaft-driver helicopter, i.e., assuming a fixed disc loading and solidity, the resulting point design, if not optimum can only be conservative.

4.1.2.4 Heavy-Lift Jet-Flap Helicopter

A range of three gross weights, three rotor diameters, and three solidities resulting in a 27-point matrix was employed in each sizing study. Due to the rotor blade duct sizes required to accommodate the mass flow, a three-bladed rotor was selected as being the most practical. Based on the results of the preliminary study, a fixed diameter of 145 feet has been selected for use in the current study. A 2-bladed jet-flap HLH was sized to determine the effect of number of blades.

4.1.3 Engine Sizing

4.1.3.1 High-Speed Helicopter

Engine size was based on the requirement in hover for an out-of-ground effect rotor thrust/gross weight ratio, $T/W = 1.0$, at 6,000 feet, 95°F using the LTVAC Rotor Performance Program Program #3 described in Appendix A, Volume II. It should be recognized that this requirement results in a rather oversize engine for the design mission hover condition at sea level, 59°F. However, the desired high speed in cruise, $V = 200$ knots, justifies this engine size. For the jet-flap rotor there exists a minimum blade duct flow area to deliver a specific rotor shaft horsepower, as explained in Section 3-1. Thus, for specific rotor diameters and number of blades, the minimum solidity is limited by the minimum blade cord required to accommodate the duct flow area. A side study proved that, for this

range of solidity, the minimum gross weight vehicle capable of meeting the mission increased as solidity increased. Thus, by using the solidity defined by the duct area limit for each diameter, the lowest permissible optimum gross weight was obtained.

4.1.3.2 Heavy-Lift Helicopter

The engine sizes for the heavy-lift shaft-driven and jet-flap helicopters are sized in the same manner as the high-speed helicopters but employ a rotor hover thrust/weight ratio = 1.1 at sea level standard conditions.

4.1.4 Engine Selection - Hot - Warm Cycle

4.1.4.1 High-Speed Jet-Flap Helicopter

Primarily because of the additional energy transformation required, the helicopter with a cold cycle propulsion system is heavier than hot or warm cycle vehicles designed for the same mission. Thus, in the interest of determining the optimum system, the sensitivity study was conducted with hot or warm cycle engines.

A limited parametric study involving warm and hot cycle engines was conducted to assist in determining the lightest weight vehicle to be used as a basis for the high-speed helicopter sensitivity study. The most optimistic (100 percent thrust recovery and 5 percent static duct pressure loss), intermediate (90 percent thrust recovery and 15 percent duct loss), and pessimistic (80 percent thrust recovery and 25 percent duct loss) cases were employed in the HSH mission routine with high, intermediate and low engine pressure ratios (engines 1, 4 and 6). Results of these studies are summarized in Figure 4-1, where minimum gross weight for each case is shown as a function of pressure ratio. It may be noted that in all cases, the

HIGH-SPEED JET-FLAP HELICOPTER
TAKEOFF GROSS WEIGHT VS PRESSURE RATIO

- TR=100% ΔP=5% } DIA=72.2 FT
- △ TR=90% ΔP=15% }
- TR=80% ΔP=25% }
- ▲ TR=90% ΔP=15% DIA=65 FT
- △ TR=90% ΔP=15% DIA=55 FT

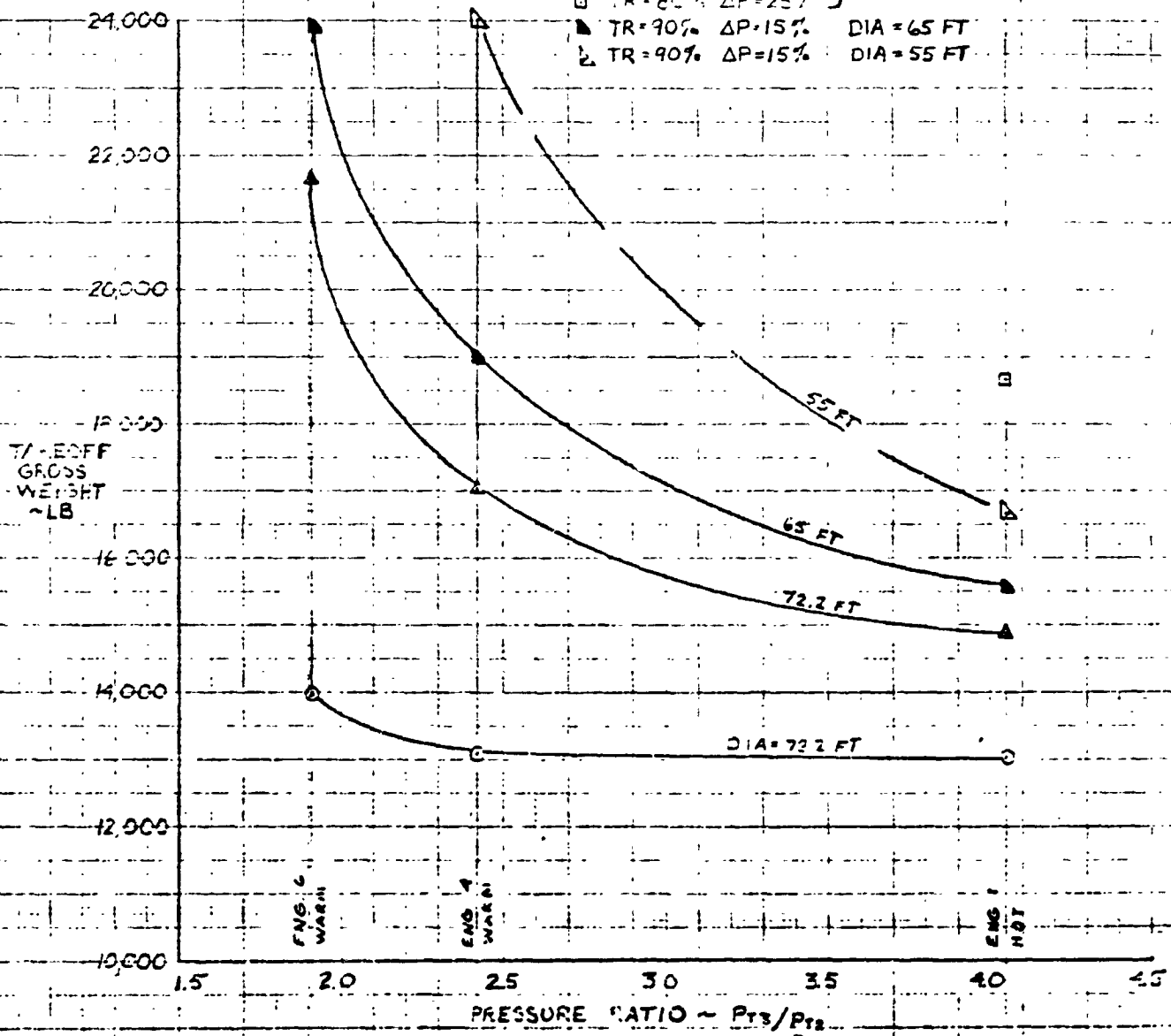


Figure 4-1

minimum gross weight occurs at the highest pressure ratio considered. For the "clean" propulsion systems (i.e., high thrust recovery and low pressure loss), the variation of minimum gross weight with pressure ratio is relatively low.

4.1.4.2 Heavy-Lift Jet-Flap Helicopter

A study of the effect on gross weight of the heavy-lift helicopter engine pressure ratio indicated that within the range of pressure ratios studied, minimum gross weight is achieved with the hot cycle pressure ratio = 4.05. This is illustrated in Figure 4-2 where gross weight is presented as a function of pressure ratio for a tip speed of 700fps. As in the case of the high-speed helicopter, the major factors contributing to this result are the lower solidity resulting from the higher pressure ratio and the more favorable power-to-weight ratio of the hot cycle.

4.1.5 Blade Drag Coefficient Sensitivity

Three relationships were considered in evaluating the sensitivity of the high-speed jet-flap helicopter gross weight with rotor profile drag.

$$1. C_D = C_{D_0} + 0.008 \bar{C}_L^2 \text{ (employed in preliminary study)}$$

$$2. C_D = C_{D_0} + 0.004 \bar{C}_L$$

$$3. C_D = C_{D_0}$$

For a jet-flap rotor blade with a 5 percent deflecting flap over the outer 30 percent radius, the most probable relationship lies between cases 2 and 3. The resulting percent change in minimum gross weight referenced to case 1 is as follows:

1

TAKEOFF GROSS WEIGHT & PRESSURE RATIO HLH ENGINE SELECTION

- TR=100% ΔP=5%
- △ TR=90% ΔP=15%
- TR=80% ΔP=25%
- DIA=145 FT
- V_{tip}=700 FT/SEC

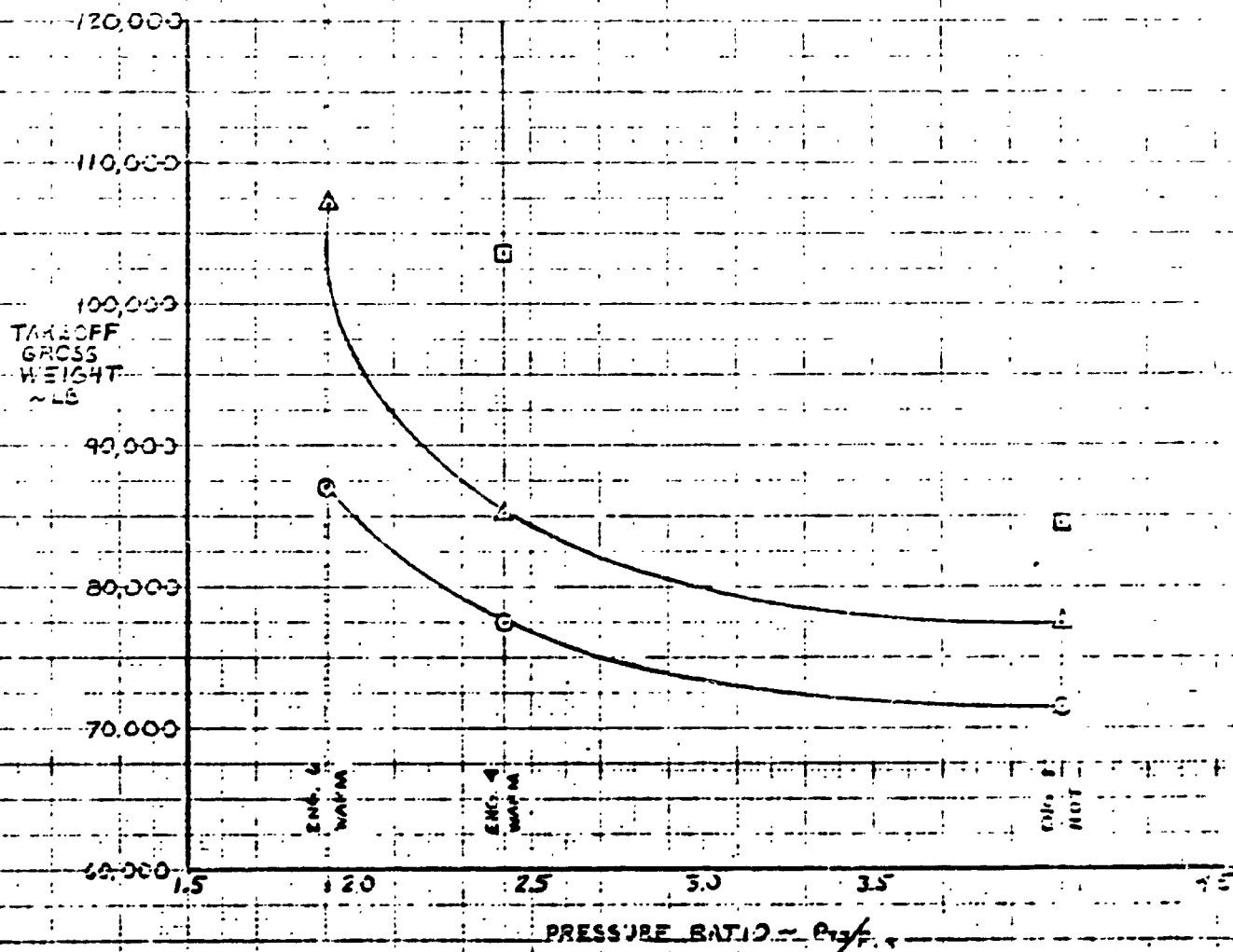


Figure 4.2

Case	Percent Gross Weight Change
1	0
2	0
3	-6.5

For the basic sensitivity study, LTV employed the limiting, most optimistic case number 3 in predicting power required.

4.1.6 Effect of Disc Loading

To determine the effect of the disc loading, a brief parametric study was conducted for the intermediate propulsion case (90 percent thrust recovery and 15 percent duct loss) at diameters of 55, 65, and 72.2 feet. Results are presented in Figure 4-3 as gross weight, and disc loading as a function of diameter. It is observed that take-off gross weight decreases as rotor diameter increases. For helicopters of this weight class, a rule-of-thumb measure of minimum disc loading is 4 psf. Since the 72.2 ft diameter produced a disc loading only slightly less than 4., it was employed for succeeding studies.

4.1.7 Rotor Duct Area/Section Area Sensitivity

Figure 4-4 presents the sensitivity of minimum gross weight due to duct area/profile area ratio. The ratio of available duct area ranges from 0.38 (original study) to 0.75. The present point design employs a duct area ratio equal to 0.5. This value was also used in the basic sensitivity study. It is observed that for this range the sensitivity is rather low (-2.75 to +2.7 percent of gross weight).

4.2 Point Design Selection

Presented in Table 4-1 is a summary of the high-speed and heavy-lift helicopters employing LTV's best estimate of the critical performance/

HIGH-SPEED HELICOPTER TAKE-OFF GROSS WEIGHT vs ROTOR DIAMETER

TR = 90%

DL = 15%

f = 12 FT²

ENGINE NO. 1

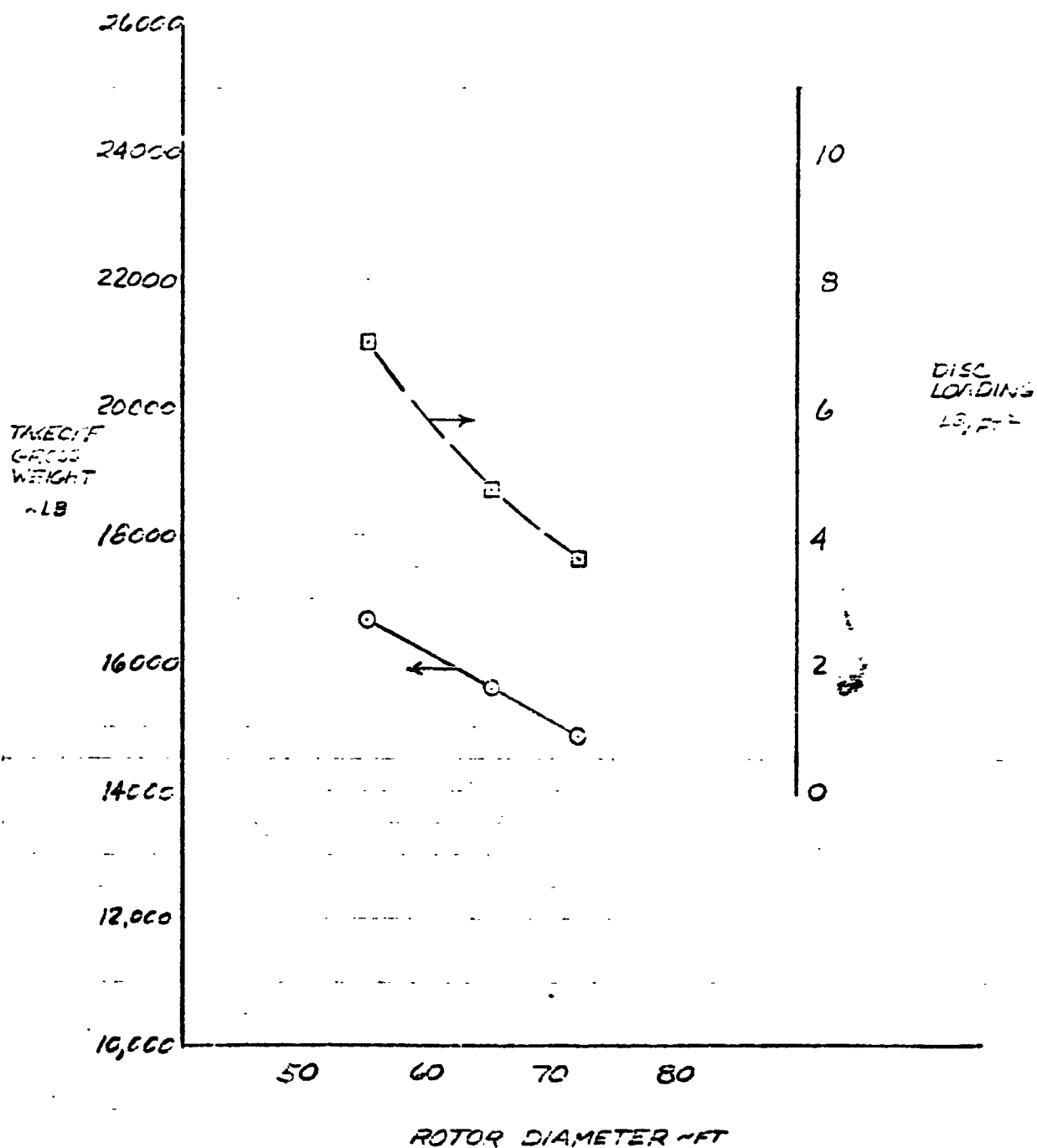


Figure 4-3

HIGH-SPEED JET-FLAP HELICOPTER
 DUCT AREA SENSITIVITY
 ENGINE NO. 1
 TR=90%, $\Delta P=15\%$

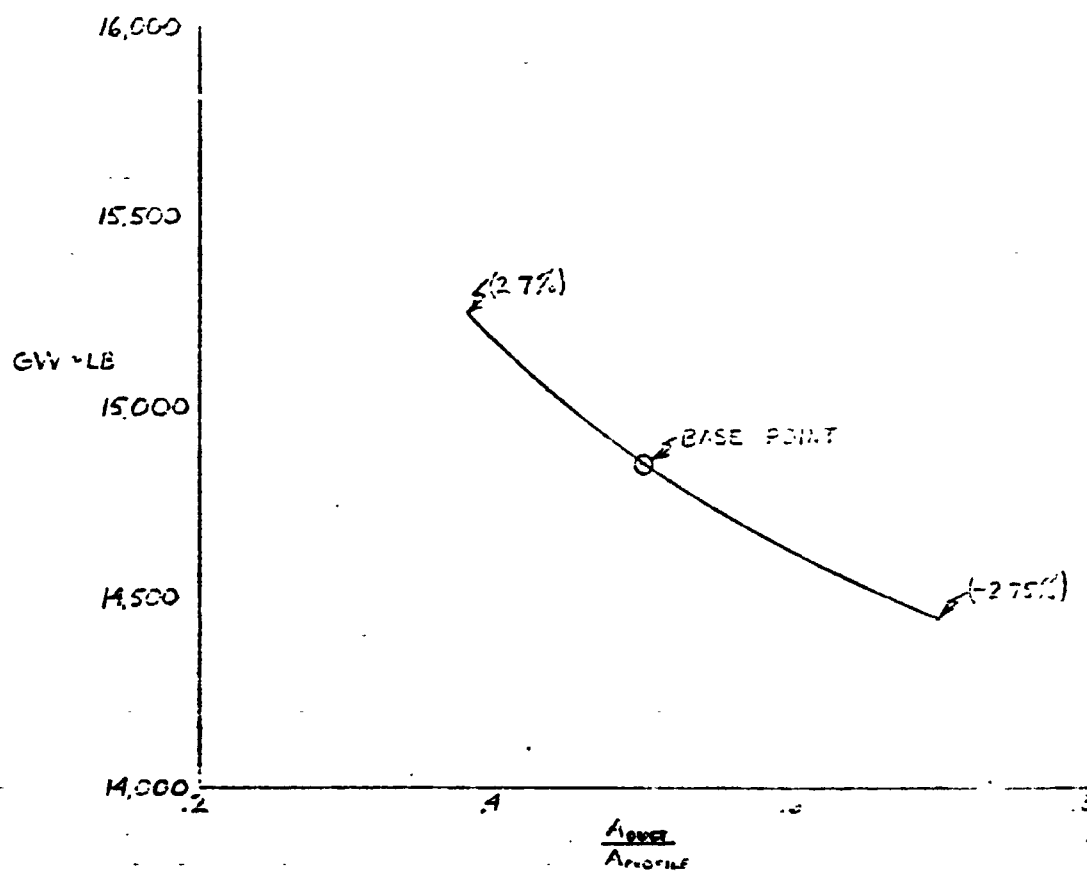


Figure 4-4

Table 4-1 Point Design Configurations

Type	Gross Weight lb	Rotor Diameter ft	Solidity	No. of Blades	Tipspeed fps	$\bar{C}_L$	No. of Engines	Rated HP/Eng	Fuel lb	Payload lb	OWE lb
High-Speed Shaft	14,870	67.0	0.0785	2	640	0.3480	2	1,200	1,760	4,000	9,100
Heavy-Lift Shaft	87,400	104.0	0.1060	6	700	0.5500	2	7,520	2,970	40,000	44,430
High-Speed Jet-Flap (Hot Cycle)	14,850	72.2	0.0340	2	640	0.6900	2	F_N 1,223	4,400	4,000	6,450
Heavy-Lift Jet-Flap (Hot Cycle)	77,400	145.0	0.0435	3	700	0.581	4	F_N 2,660	3,400	40,000	34,000
High-Speed Jet-Flap (Cold Cycle)	16,000	72.2	0.0375	2	640	0.671	2	3,125	4,630	4,000	7,370
Heavy-Lift Jet-Flap (Cold Cycle)	83,200	145.0	0.0465	3	700	0.584	4	6,480	3,900	40,000	39,300

propulsion parameters; i.e., thrust recovery, duct loss, equivalent flat plate drag, engine pressure ratio, etc. Additional information on the point designs is tabulated in Tables 4-2 and 4-3.

4.2.1 High-Speed Shaft-Driven Helicopter

The rotor diameter equal to 67 feet was observed to be optimum throughout the range of solidities. It was determined that a rather large rotor diameter was required in order to delay the upper stall limit, permitting the high cruise speed. The accompanying solidity was determined by a maximum coning angle of 2.5 degrees.

4.2.2 High-Speed Jet-Flap Helicopter (Hot Cycle)

As shown in Table 4-1, the $\bar{C}_L$ of the optimum high-speed jet-flap helicopters is substantially below the achievable value. It was found advantageous to reduce the design $\bar{C}_L$ by increasing the rotor diameter in order to conserve on engine size and its associated high fuel requirement throughout the mission.

4.2.3 Heavy-Lift Jet-Flap Helicopter (Hot Cycle)

The tip speed of the heavy-lift jet-flap helicopters was fixed at $V_T = 700\text{fps}$ to be consistent with the shaft-driven machine. For this phase of the study, the diameter was held constant at $DIA = 145$ feet, which was indicated as optimum in the preliminary study.

4.2.4 High-Speed Jet-Flap Helicopter (Cold Cycle)

The engine selection for the cold cycle high-speed helicopter is presented in Figure 3-15 indicating a compressor pressure ratio equal to 4.2 as the optimum. As noted in the figure, the following values are fixed: thrust recovery = 90%, duct loss = 15%, diameter = 72.2 ft, and

TABLE 4-2 HSI POINT DESIGN CHARACTERISTICS

ENGINES RATED AT 6,000 FEET 95°F

107MAN 5 MINUTES AT ORIGIN, 10 MINUTES AT MID MISSION

High-Speed Helicopter
Payload 4,000 Pounds
200 Knots Cruise Speed
450 Fuel/air Miles R/A

	Gross Wt. lb.	Empty Wt. lb.	OWE lb.	Net Wt. lb.	Unit Wt. lb.	Motor Wt. lb.	Disc Loading per sq. ft.	Efficiency	Ground Wt. lb.	Empty Wt. lb.	Pressure Ratio	Cycle	Thrust Recovery	Duct Pressure	Parasite Drag Eq. ft.	Tip Speed Fps	Area Ratio	Remarks	Operating Cost \$/Ton hr
1	24,100	5,512	6,060	4,048	3,814	72.2	3.45	0.0300	1,700	0.392	4.05	HOT	0.90	0.05	12.0	640	0.50		0.639
2	14,350	5,926	4,474	4,376	6,455	72.2	3.63	0.0340	1,930	0.399	4.05	HOT	0.90	0.15	12.0	640	0.50		0.813
3	13,550	6,202	6,750	4,800	7,340	72.2	3.80	0.0385	2,180	0.399	4.05	HOT	0.90	0.25	12.0	640	0.50		0.921
4	10,950	5,734	7,280	5,510	9,134	72.2	4.14	0.0405	2,300	0.397	4.05	HOT	0.80	0.15	12.0	640	0.50		0.846
5	10,500	5,352	5,900	5,600	4,510	72.2	3.30	0.0295	1,670	0.397	4.05	HOT	1.00	0.15	12.0	640	0.50		0.792
6	12,950	5,204	5,750	5,800	5,372	72.2	3.17	0.0315	1,785	0.401	4.05	HOT	0.90	0.15	8.0	640	0.50		0.777
7	15,500	5,392	6,440	5,560	7,498	72.2	4.05	0.03700	2,100	0.388	4.05	HOT	0.90	0.15	16.0	640	0.50		0.844
8	14,050	5,926	6,474	4,376	6,455	72.2	3.63	0.0340	1,930	0.399	4.05	HOT	0.90	0.15	12.0	640	0.50		0.674
9	14,850	5,926	6,474	4,376	6,455	72.2	3.63	0.0340	1,930	0.399	4.05	HOT	0.90	0.15	12.0	640	0.50		0.953
10	11,950	4,627	5,175	2,775	4,092	67.0	3.39	0.0230	1,210	0.387	4.05	HOT	0.98/0.92	$R_p = 1.0$ 0.0955	9.4	750/680	0.60	Desired mg = 0.4	0.779
11	12,830	5,832	6,380	2,450	3,528	67.0	3.64	0.0435	2,450	0.454	1.91	WARM	0.98/0.92	$R_p = 1.2$ 0.098	8.0	750/640	0.75	Desired mg = 0.45	0.737
12	21,650	10,152	10,700	6,950	10,923	72.2	5.29	0.0820	4,650	0.468	1.91	WARM	0.90	0.15	12.0	640	0.50		
13	14,000	6,452	7,000	3,000	3,348	72.2	3.42	0.0580	3,290	0.461	1.91	WARM	1.00	0.05	12.0	640	0.50		
14	17,000	7,552	8,200	4,800	6,512	72.2	4.16	0.0620	3,510	0.450	2.42	WARM	0.90	0.15	12.0	640	0.50		
15	13,100	5,552	6,100	3,000	3,458	72.2	3.20	0.0430	2,440	0.424	2.42	WARM	1.00	0.05	12.0	640	0.50		
16	13,000	5,102	5,650	3,350	4,409	72.2	3.18	0.0260	1,470	0.393	4.05	HOT	1.00	0.05	12.0	640	0.50		
17	18,700	7,582	8,130	6,570	11,352	72.2	4.57	0.0470	2,660	0.406	1.05	HOT	0.80	0.25	12.0	640	0.50		
18	16,660	5,672	7,260	5,480	10,138	55.0	7.02	0.0970	2,460	0.399	4.05	HOT	0.90	0.15	12.0	640	0.50		
19	14,000	16,442	10,590	9,090	13,960	55.0	10.10	0.1160	3,090	0.434	2.42	WARM	0.90	0.15	12.0	640	0.50		

4-15/16

Table 4-2 HSH Point Design Characteristics (Continued)
 ENGINES SIZED AT 6,000 FEET 95°F
 HOVER 9 MINUTES AT ORIGIN, 10 MINUTES AT MID MISSION

High-Speed Helicopter Payload 4,000 Pounds 200 Knots Cruise Speed 150 Nautical Miles R/A																			
	Gross Wt. lb.	Empty Wt. lb.	OKE lb.	Fuel lb.	Installed GHP HP	Rotor Dia. ft.	Disc Loading p.s.f.	Solidity	Chord ft.	Empty Wt. Gross Wt.	Pressure Ratio	Cycle	Thrust Recovery	Duct Pressure	Parasite Drag Sq. Ft.	Tip Speed ft/sec	Area Ratio	Remarks	Operating Cost \$/hr - MI
20	15,500	6,302	6,850	4,750	7,980	65.0	4.70	0.0410	2.090	0.404	4.05	HOT	0.90	0.15	12.0	640	0.50		
21	16,950	8,502	9,050	5,900	14,592	65.0	5.72	0.0780	3.980	0.449	2.42	WARM	0.90	0.15	12.0	640	0.50		
22	24,000	13,233	11,780	8,220	11,239	65.0	7.24	0.1215	6.200	0.468	1.91	WARM	0.90	0.15	12.0	640	0.50		
23	17,400	8,912	7,460	5,940	9,636	72.2	4.25	0.0480	2.720	0.397	4.05	HOT	0.80	0.15	12.0	640	0.38		
24	14,440	5,692	6,240	4,200	6,090	72.2	3.53	0.0270	1.530	0.394	4.05	HOT	0.90	0.15	12.0	640	0.75		
25	27,000	12,652	13,200	9,800	14,644	72.2	6.60	0.1125	5.810	0.468	2.00	COLD	0.90	0.15	12.0	640	0.50		
26	16,700	7,202	7,750	4,950	5,689	72.2	4.08	0.0515	2.920	0.431	3.00	COLD	0.90	0.15	12.0	640	0.50		0.809
27	16,000	6,822	7,370	4,630	5,944	72.2	3.91	0.0375	2.120	0.426	4.20	COLD	0.90	0.15	12.0	640	0.50	Spill 100% Tip	
28	12,200	5,032	5,580	2,620	4,488	72.2	2.98	0.0260	1.470	0.412	4.05	HOT	0.90	0.15	12.0	640	0.50	Spill 75% Tip	
29	12,620	5,222	5,770	2,850	4,776	72.2	3.08	0.0275	1.560	0.414	4.05	HOT	0.90	0.15	12.0	640	0.50	Spill 50% Tip	0.799
30	13,075	5,397	5,945	3,130	5,095	72.2	3.20	0.0295	1.670	0.413	4.05	HOT	0.90	0.15	12.0	640	0.50	Spill 25% Tip	
31	13,600	5,602	6,150	3,450	5,491	72.2	3.32	0.0310	1.760	0.411	4.05	HOT	0.90	0.15	12.0	640	0.50	Spill 25% Tip	
32	17,000	7,092	7,600	5,400	7,840	72.2	4.16	0.0376	2.130	0.415	4.05	HOT	0.90	0.15	12.0	640	0.50	Spill 25% Tip	0.847
33	14,400	6,652	7,200	3,110	5,993	72.2	3.52	0.0336	1.900	0.461	4.05	HOT	0.90	0.15	12.0	640	0.50	Spill 25% Tip	0.791
34						72.2					4.05	HOT	0.90	0.15	12.0		0.50	Spill 25% Tip	
35	14,870	8,657	9,134	1,736	2,661	67.0	4.22	0.0785	4.130	0.585	NA	NA	NA	NA	9.6	640	NA	Spill 25% Tip	0.742
36	15,000	6,082	6,630	4,620	7,020	72.2	3.72	0.043	2.44	0.476	4.05	HOT	0.90	0.15	12.0	640	0.38		

Notes: 1) Maintenance Manhours per Flight Hour used for Operating Cost except where noted.

4-17/78

Rear-Lift Helicopter
 Payload 20 tons
 10 Nautical Miles R/A No Kolls

Table k-3 HLM Point Design Characteristics
 EXHIBITS BIZED AT SEA LEVEL T/W = 1.0
 HOVER 5 MINUTES AT ORIGIN 2 MINUTES

Item	Gross wt. lb	Empty Wt. lb	OWB lb	Fuel lb	Installed GHP HP	Motor Dia. in	Disc Loading per ft	Efficiency	Chord ft	No. of blades	Pressure Ratio	Cycle	Thrust Recovery	Duct Pressure Loss	Empty Wt. Gross Wt.	Tip Speed	Area Ratio	Remarks	Operating Cost \$/ton hr	Operating Cost \$/hr. hr.	Turny Cost \$/hr
1	87,400	43,730	44,430	2,970	15,710	104.0	10.30	0.1060	2.88	6	NA	NA	NA	NA	0.501	700	NA	Start Drive	1.241	413	1.814
2	82,500	37,480	38,180	4,320	30,335	145.0	5.00	0.0502	3.81	1	4.20	COLD	0.90	0.150	0.454	640	0.44		1.375	437	1.737
3	80,600	35,770	36,470	4,200	30,110	140.0	5.24	0.04500	3.30	3	4.20	COLD	0.90	0.150	0.443	640	0.62		1.394	442	1.907
4	84,900	39,350	40,050	5,440	39,555	145.0	5.18	0.0590	4.40	3	4.20	COLD	0.79	0.150	0.460	640	0.44		1.310	410	1.692
5	74,300	35,670	36,370	3,530	25,000	145.0	4.84	0.0460	3.49	3	4.20	COLD	1.00	0.150	0.446	640	0.44		1.376	437	1.350
6	83,000	37,200	37,900	4,900	30,120	145.0	5.02	0.0630	5.24	3	2.73	WARM	0.90	0.150	0.451	640	0.44		1.308	424	1.621
7	77,400	33,240	34,000	3,400	20,090	145.0	4.69	0.0435	5.30	3	4.05	HOT	0.90	0.150	0.430	700	0.50				
8	87,250	43,240	43,950	5,320	39,300	145.0	5.28	0.0845	6.41	3	1.91	WARM	1.00	0.050	0.441	700	0.50				
9	71,500	34,170	34,870	2,685	17,735	145.0	4.70	0.0590	4.48	3	2.42	WARM	1.00	0.050	0.399	700	0.50				
10	71,500	28,550	29,250	2,250	20,500	145.0	4.33	0.0350	2.66	3	4.05	HOT	1.00	0.050	0.406	700	0.50				
11	107,100	59,420	60,050	7,050	31,095	145.0	6.49	0.1255	9.52	3	1.91	WARM	0.90	0.150	0.554	700	0.50				
12	65,000	33,920	34,700	4,500	26,700	145.0	5.16	0.0760	5.77	3	2.42	WARM	0.90	0.150	0.469	700	0.50				
13	24,300	38,090	38,800	5,500	23,705	145.0	5.11	0.0540	4.10	3	4.05	HOT	0.80	0.250	0.451	700	0.50				
14	79,500	44,010	44,730	4,770	31,780	145.0	5.42	0.0645	4.09	3	3.00	COLD	0.90	0.150	0.491	700	0.50				
15	33,200	38,650	39,300	5,000	28,800	145.0	5.04	0.0465	3.53	3	4.20	COLD	0.90	0.150	0.465	700	0.50				
16	74,000	33,570	34,370	3,630	27,670	145.0	4.73	0.0425	3.23	3	4.05	HOT	0.90	0.150	0.432	800	0.50				
17	105,500	57,150	57,950	7,550	29,240	125.5	8.50	0.10508	3.48	6	NA	NA	NA	NA	0.542	700	NA	Long Range HLM	0.733	445	2.128
18	10,700	45,150	45,950	16,750	64,300	124.0	8.50	0.077	5.00	3	4.05	HOT	0.90	0.150	0.441	700	0.50	Long Range HLM	0.865	325	2.125
19	75,700	32,670	33,370	2,330	16,280	145.0	4.99	0.0485	5.52	2	1.91	WARM	0.98	0.041	0.431	720	0.75	Desired	1.166	378	1.299
20	66,200	23,540	24,230	1,970	19,800	145.0	4.01	0.0245	2.79	2	4.05	HOT	0.98	0.056	0.355	720	0.60	Desired	1.178	382	1.666
21	103,500	53,780	54,500	9,000	48,730	145.0	6.27	0.1100	8.35	3	2.42	WARM	0.80	0.250	0.520	700	0.50				
22	91,800	40,000	40,800	11,000	30,900	117.3	8.50	0.0825	7.58	2	1.91	WARM	0.98	0.041	0.436	720	0.75	Desired	0.771	468	1.921
23	72,400	28,750	29,500	2,900	24,830	145	4.38	0.033	3.76	2	4.05	HOT	0.90	0.15	0.398	700	0.5	Desired	1.247	468	1.971

k-19/20

Notes: Maintenance loadings per Flight Hour of 18 used for Operating Cost except where noted

tipspeed = 700 fps. The much higher gross weight at a pressure ratio of 2 is due to the large engine size and larger solidity contributing to higher fuel requirement accompanied by reduced fuel available.

4.2.5 Heavy-Lift Jet-Flap Helicopter (Cold Cycle)

Table 4-4 presents the results of a pressure ratio selection for the heavy-lift helicopter. For this study, pressure ratios of 3.0 and 4.2 were investigated. Here, as with the previous engine selection studies, the higher pressure ratio engine contributed to the lower gross weight helicopter.

TABLE 4-4 Cold Cycle Heavy-Lift Helicopter

TR	= 90%	
DL	= 15%	
DIA - ft	= 145	
No. of Blades	= 3	
No. of Engines	= 4	
Pressure Ratio	3.0	4.2
Gross Weight lb	89,500	83,200
Solidity	0.0645	0.0465
SHP _{Rated} Engine hp	7,150	6,480

4.2.6 High-Speed Shaft-Driven Helicopter Velocity Sensitivity

The sensitivity of cruise speed on the high-speed shaft-driven helicopter point design gross weight was determined. Holding the geometric rotor parameters of the original point design constant, the cruise speed was varied from 120 knots to 200 knots. As shown in Figure 4-5, the minimum gross weight, GW = 14,360 pounds, occurred near a cruise speed of 150 knots.

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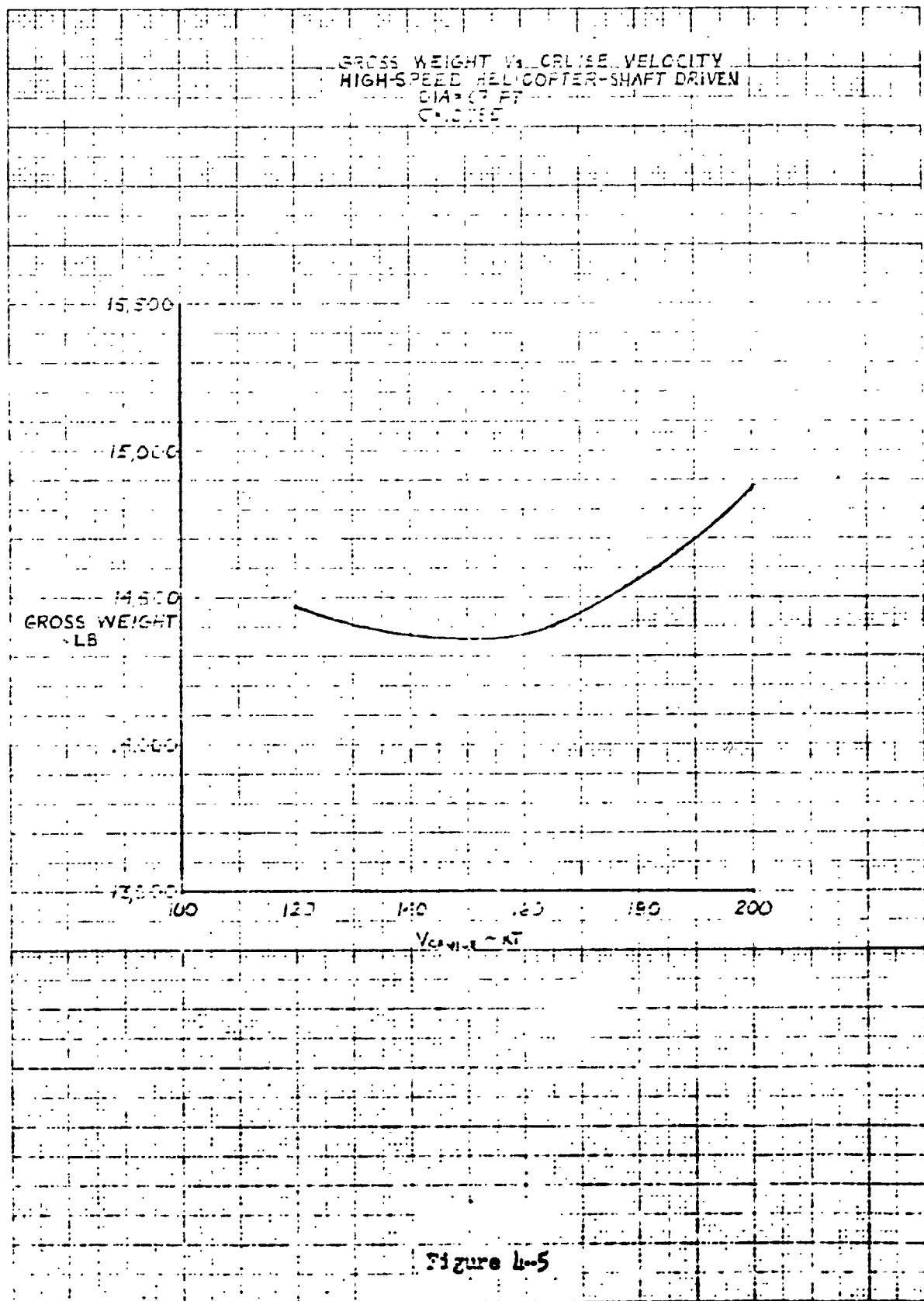


Figure 4-5

This represents a gross weight reduction of 3.5 percent from the original point design at $V = 200$ knots.

4.2.7 Azimuth Survey

In order to identify possible speed limitations on the high-speed jet-flap helicopter, a rotor azimuth survey has been conducted employing LTVAC Rotor Performance Program No. 2. This study, neglecting engine power limits, considers the rotor local $c_l/c_{l_{max}}$ and local flap deflection over a range of velocities bracketing the power limit speed shown in Table 4-8. The greater blowing (and higher $c_{l_{max}}$) at the highest airspeed contributes to a reduction in local $c_l/c_{l_{max}}$. Thus, although the local flap deflection increases with speed, the local $c_l/c_{l_{max}}$ becomes less severe over this range of velocity (see Figure 4-6).

4.2.8 Tipspeed Effect (Heavy-Lift)

Negligible difference in gross weight is realized in increasing the tipspeed to 800 fps. This indicated in Table 4-5.

4.2.9 Dorand Design Estimates

Several additional side studies of both heavy-lift and high-speed helicopters have been completed. These studies represent the estimates by Giravions-Dorand engineers of basic study parameters detailed in Table 4-6. The four jet-flap configurations include:

1. Pure jet-flap high-speed helicopter (hot cycle engine)
2. Jet-flap plus auxiliary direct thrust power split, high-speed helicopter (warm cycle engine)

HIGH-SPEED JET-FLAP HELICOPTER AZIMUTH SURVEY

1

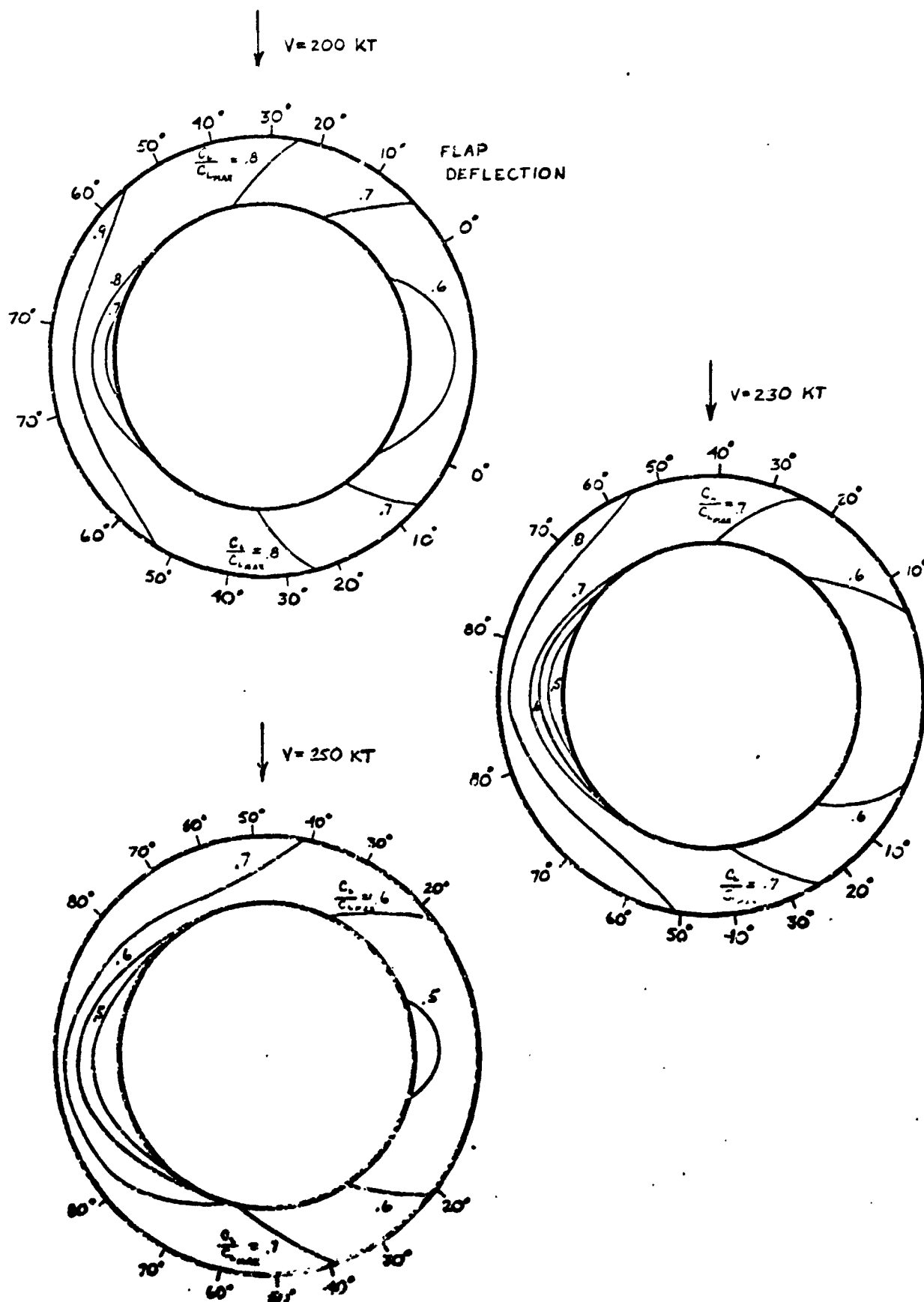


Figure 4-6

Table 4-5 Heavy-Lift Helicopter
Tip Speed Effect

TR	= 90%		
DL	= 15%		
Diameter	= 145 ft.		
No. of Blades	= 3		
No. of Engines	= 4		
Engine No.	= 1		
Tip speed	fps	700*	800
Gross Weight	lb	77,400	78,000
Solidity		0.0435	0.0425
$FN_{Rated}/Engine$	lb	2,660	2,620

*Basic Point Design

Table 4-6 Dorand Inputs - High-Speed Helicopter

Mission

Engine sizing: 6,000 feet 95°F
 R/A: 150 nautical miles
 Vcruise: 200 knots
 Payload: 4,000 pounds

Pure Helicopter Parameters

GW: 11,950 pounds
 Diameter: 67 feet
 Flat plate drag factor = 9.4 square feet
 No. of blades: 2
 No. of engines: 2
 Hover: $V_T = 720$ ft/sec
 TR = 0.98
 Cruise: $V_T = 680$ ft/sec
 TR = 0.92
 Engine pressure ratio: 4.05
 Duct loss coefficient = 1.0 ($\frac{\Delta P}{P} = 0.0955$)
 Duct MN = 0.4
 Solidity = 0.023
 Rated thrust per engine: 775 pounds

Power Split-Jet-Flap Rotor + Direct Thrust

Diameter: 67 feet
 Flat plate drag factor = 8 square feet
 No. of blades: 2
 No. of engines: 2
 Hover: $V_T = 720$ ft/sec
 TR = 0.98
 Cruise: $V_T = 640$ ft/sec
 TR = 0.92
 Engine pressure ratio: 1.91
 Duct loss coefficient = 1.2 ($\frac{\Delta P}{P} = 0.098$)
 Duct MN = 0.45

	<u>Dorand Power Split</u>	<u>LTV Power Split</u>
GW:	13,900 pounds	12,830 pounds
Solidity	0.0457	.0435
Rated Thrust per Engine:	1,562 pounds	1,400 pounds

Table 4-6 Dorand Inputs - Heavy-Lift Helicopter
(Continued)

Mission

Engine sizing: $T/W = 1.1$, SL Std.
R/A: 10 NM
Vcruise: 40 knocs
Payload: 4000 lbs.
Hover 5 min. at origin, 2 min. at mid-mission

Heavy-Lift Warm Cycle

GW: 75,700 lbs.
Dia: 145 ft.
No. of blades: 2
No. of engines: 4
 $V_T = 720$ fps
Thrust recovery = .98
Engine pressure ratio: 1.91
Duct loss coefficient = 0.70
Duct MN = 0.3
Solidity = .0485
Rated thrust per engine = 3230 lb

Heavy-Lift Hot Cycle

GW: 66,200 lb
Dia = 145 ft.
No. of blades = 3
No. of engines = 4
 $V_T = 720$ fps
Thrust recovery = .98
Engine pressure ratio = 4.05
Duct loss coefficient = 1.0
Duct MN = 0.3
Solidity = .0245
Rated thrust per engine = 1875 lb

3. Heavy-lift helicopter (warm cycle)

4. Heavy-lift helicopter (hot cycle)

The percent flow split recommended by the Dorand team resulted in excessive airflow to the flap. Using the same parameters but employing a flow split where only enough thrust to overcome the parasite drag of the vehicle plus ram drag was diverted to direct thrust resulted in a substantial weight decrease.

4.3 Sensitivity Study

The purpose of this study is to establish the sensitivity of the high-speed jet-flap helicopter gross weight with variation in certain key performance parameters. These parameters include thrust recovery factor, duct loss, and parasite drag. Prior to conducting the sensitivity study with respect to the above, additional related side studies, previously described, were required to determine:

- . Choice of blade drag coefficient
- . Choice of engine
- . Justification of diameter selection
- . Rotor $\frac{\text{Duct Area}}{\text{Section Area}}$ Sensitivity
- . Engine cycle effect on gross weight

4.3.1 Basic Sensitivity Study

Upon completion of the previously mentioned supporting studies, the following parameters were fixed throughout the basic sensitivity study:

Engine cycle	= hot
Engine pressure ratio	= 4.05
Rotor diameter	= 72.2 feet

Rotor duct area/section area = 0.5

Rotor profile drag

coefficient, C_d = C_{D_0}

The base case for the sensitivity study employs LTV's best estimate for the values of the three parameters shown below in Table 4-7.

Table 4-7 Sensitivity Study Parameters Range

Parameter	LTV Best Estimate	Calculation Range
Thrust Recovery	90%	80% to 100%
Duct Pressure Loss	15%	5% to 25%
Equivalent Flat Plate		
Drag Area, Ft^2	12	8 to 16

Figure 4-7 is a "nomograph type" presentation of the high-speed sensitivity study. An arbitrary example case is indicated

4.4 Compound Thrust Systems

4.4.1 Tip-Jet/Jet Flap

The effect on the point design of splitting the gas flow between the flap and tip-jet nozzle was investigated for both the high speed and heavy lift missions. Figure 4-8 presents the resulting minimum gross weights capable of completing the design high-speed mission as a function of the percent ESHP supplied by the tip jet. Also shown is the rotor average lift coefficient, $\bar{C}_L$, at 6,000 feet 95°F and sea level standard. Engine No. 1 is utilized in this study as in other high-speed side studies.

Figure 4-9 presents the results of the tip-jet power split for the heavy-lift helicopter. At low percent flow to the tip, there is only a slight decrease in fuel required when compared to the pure helicopter.

HIGH-SPEED HELICOPTER SENSITIVITY STUDY

BASE POINT: TR=90%, DL=15%, f=12 ft², GW=14850 LB

EXAMPLE CASE

TR=86%
DL=17%
f = 13 ft²
Δ%GW = 9%
GW = 109 × 14850
= 16200 LB

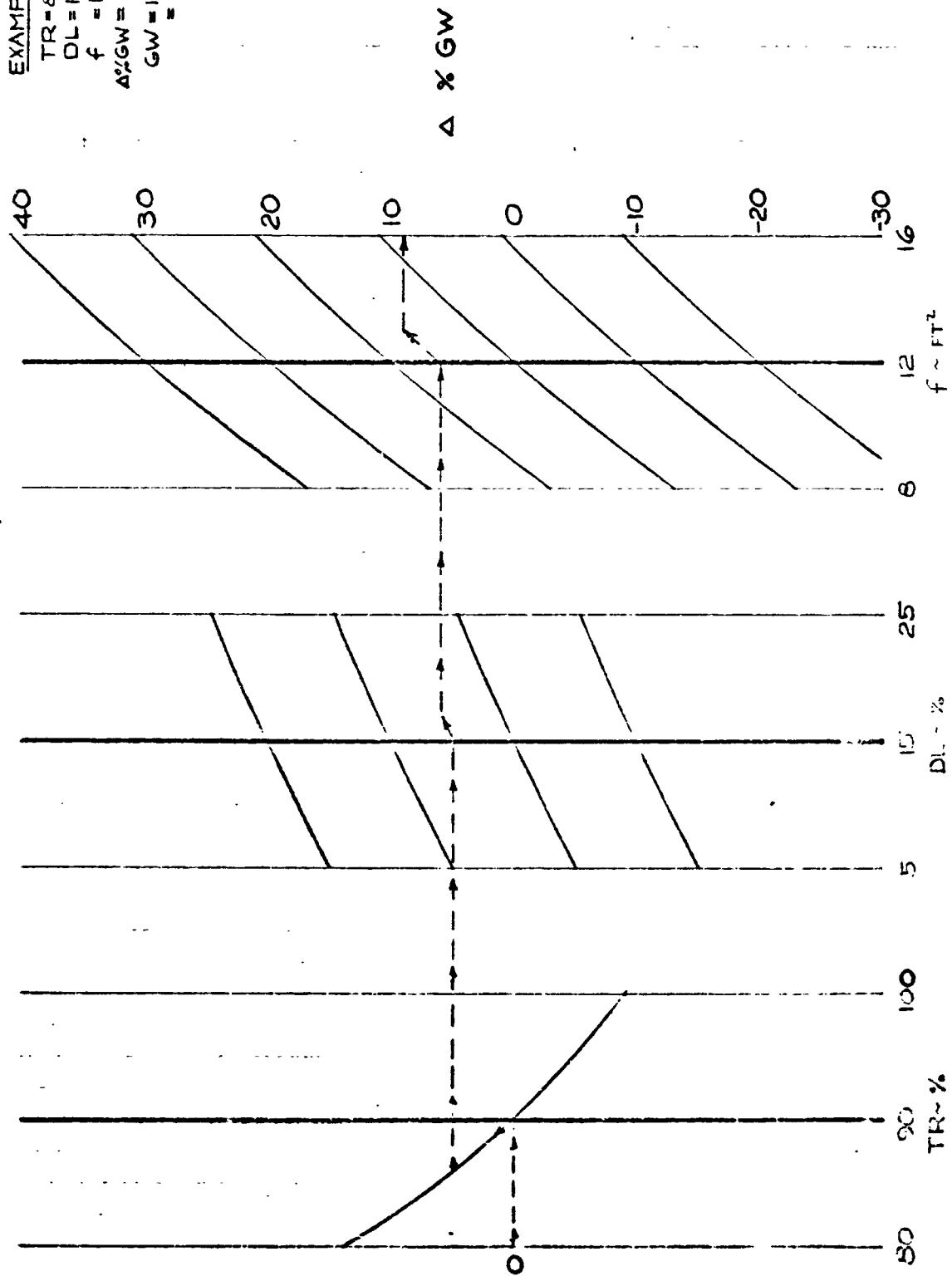


Figure 4-7

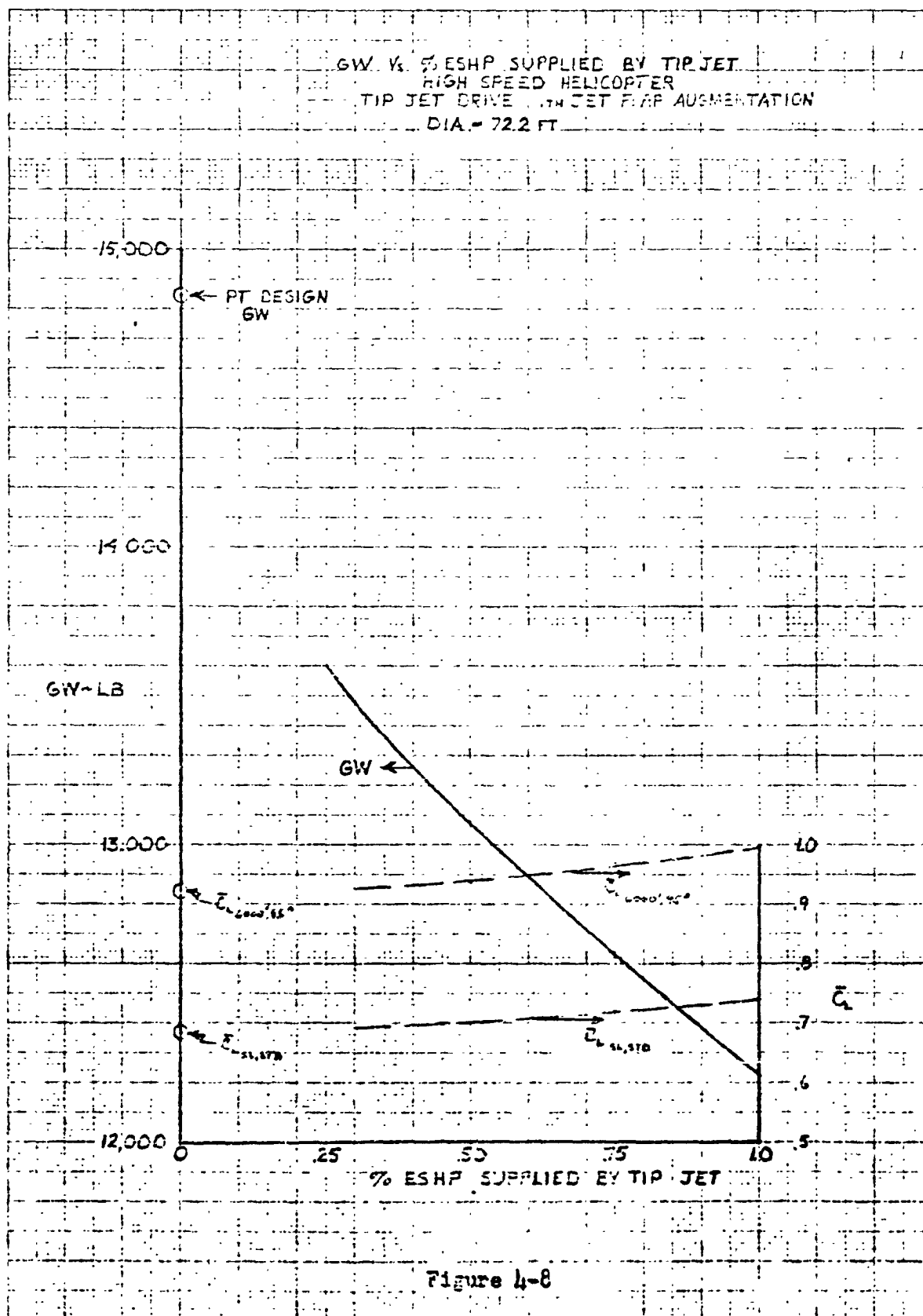


Figure 4-8

7-21

GW vs % ESHR SUPPLIED BY TIP NOZZLE
HEAVY LIFT HELICOPTER

ENG. PR.=4.05

$V_T = 700 \text{ fps}$

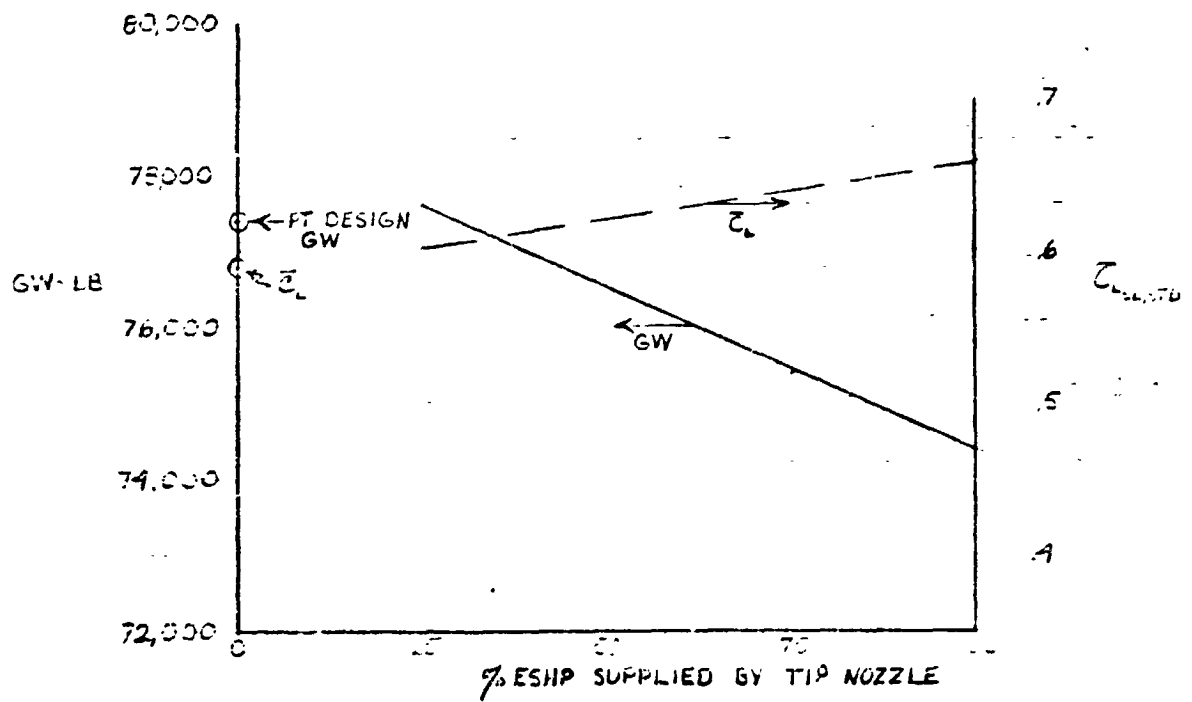


Figure 4-9

However, a substantial decrease in fuel available occurs due to the installation of the heavy-tip nozzles, thus contributing to the higher takeoff gross weight. For further increases in percent flow to the tip, the gross weight decreases, due to decreasing engine size. Of course, depending on control requirements, a minimum airflow to the flap would be required.

4.4.2 Auxiliary Thrust

The high-speed helicopter has been sized for three cases using auxiliary cruise thrust; i.e., (1) a turbojet exhaust, (2) a turbojet fan, and (3) a turboprop. At the design cruise speed of 200 knots, sufficient power is supplied by the thruster to overcome the parasite drag and engine ram drag. For the jet and fan types of cruise thrusters, the engine size for each parametric point is based on the jet-flap hover requirement only. In addition to sizing engines to drive the rotor in hover, the turboprop configuration includes a separate turboshaft engine rated at 1320 shaft horsepower. This auxiliary engine powers an 11.3-foot diameter tail propeller in the cruise mode. The following basic high-speed parameters were fixed:

Engine selected	= No. 1 (PR = 4.05)
Thrust recovery	= 90%
Duct loss	= 15%
Equivalent flat plate area	= 12 ft ²
Duct area ratio	= 0.5

The resulting minimum gross weights for each case are shown in Table 4-8. The turbojet mode using a pressure ratio of 4.05 at this cruise speed is less efficient than the pure rotor for developing forward thrust and thus results in a higher gross weight. However, the fan with a lower exhaust velocity than the turbojet contributes to a lower overall gross weight (including the fan weight penalty) than the pure helicopter.

Table 4-8 Auxiliary Thrust Point Designs

	Pure Helicopter	Turbojet	Turbofan	Turboprop
Gross Weight lb	14,850	17,000	14,400	18,800
$F_{n_{Rated}}/Engine$ lb	1,222	1,465	1,135	1,640
No. of Engines	2	2	2	2
No. of Blades	2	2	2	2
Rotor Solidity	0.034	0.0376	0.0336	0.0405
Rotor Diameter ft	72.2	72.2	72.2	72.2
Cruise Speed knots	200	200	200	200
NRP Speed knots	200	200	224	200
MRP Speed knots	203	205	230	217

The incompatibility of a single propulsion system powering both a jet-flap rotor and a shaft-driven propeller necessitated a separate engine to drive the propeller. The accompanying weight penalties lead to a significantly greater gross weight for the turboprop case.

Supplying the power to overcome parasite and ram drag by means of an auxiliary thruster eliminates the propulsive thrust requirement on the rotor. Azimuth surveys of local $c_l/c_{l_{max}}$ and local flap deflection presented in Figure 4-10 show the effect on a jet-flap rotor of blowing only enough to meet the profile and induced power requirements. The limited blowing results in a loss of jet-flap effectiveness thus increasing the flap deflections required to maintain a trimmed condition. A comparison of Figures 4-6 and 4-10 illustrates the effects of reduced blowing on the performance of a jet-flap rotor in high speed flight.

HIGH-SPEED JET-FLAP HELICOPTER WITH AUXILIARY THRUST AZIMUTH SURVEY

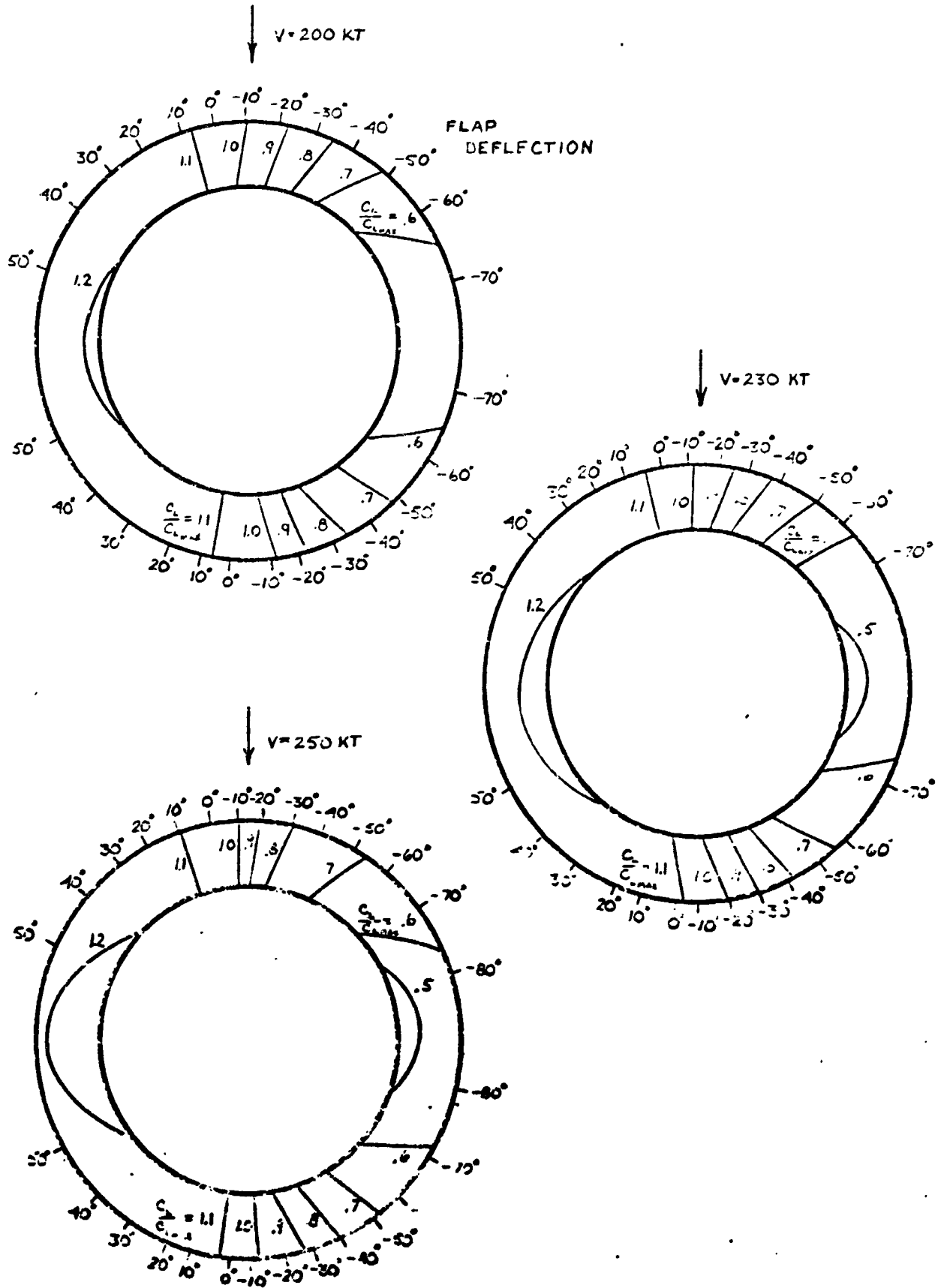


Figure 4-10.

5.0 OTHER JET-FLAP APPLICATIONS

In addition to the basic high-speed and heavy-lift missions used in the parametric comparison studies, some other applications of the jet-flap rotor were briefly examined. As a pure helicopter performing a long range heavy-lift mission two jet-flap helicopters were sized and were compared with the shaft-driven helicopter performing the same mission. Results of this study are discussed below. Another pure helicopter application examined by Giravions Dorand for the French DOD is a stationary radar surveillance platform. This is also discussed in paragraph 5.1 below. Another promising concept of the jet-flap rotor is a composite stoppable rotor aircraft. Among the potential applications of this concept are the Army surveillance mission, air rescue mission, and the close support mission. The surveillance and air rescue vehicles have been sized to perform typical missions.

5.1 LONG RANGE HEAVY-LIFT HELICOPTER

Heavy-lift shaft and jet-flap helicopters were sized to meet the following long range mission.

Payload	= 40,000 lb
Range	= 100 NM
Cruise speed with payload	= 70 knots
Cruise speed without payload	= 125 knots
Power requirement	= $T/W = 1.06$ at 4,000 foot 95°Y
Warm-up allowance	= 2 min at MFP
Hover with payload	= 10 min
Hover minus payload	= 20 min
Reserve	= 10% initial value

The shaft-driven helicopter was designed on the basis of present-day heavy-lift helicopter characteristics previously discussed. The best estimates previously derived were employed for the jet-flap parameters. In addition a configuration based on Dorand's estimates is included. Table 5-1 presents the results for the heavy-lift long range mission.

It may be noted from Table 5-1 that both jet-flap configurations are lighter than the shaft-driven helicopter but operating cost is slightly higher.

Table 5-1

Long Range HLH Characteristics

	<u>Shaft-Driven</u>	<u>LTV Estimate</u>	<u>Dorand Estimate</u>
Cycle	-----	Hot	Warm
Pressure ratio	-----	2.05	1.91
Thrust recovery	-----	0.90	0.98
Duct pressure loss	-----	0.15	0.041 ($K_t=0.7$)
Tip speed, fps	700	700	720
Duct area ratio	-----	0.5	0.75
No. of blades	6	3	2
Gross Weight,	105,500	102,700	91,800
Empty Weight, lb	57,150	45,150	40,000
OWE, lb	57,950	45,950	40,500
Fuel Weight, lb	7,550	16,750	11,000
Installed GHP	25,240	64,300	30,900
Disc loading, psf	8.5	8.3	3.5
Rotor diameter, ft	125.5	124	117.3
Solidity	0.1058	0.077	0.0825

Table 5-1 (Concluded)

Long Range HLH Characteristics			
	<u>Shaft-Driven</u>	<u>LTV Estimate</u>	<u>Dorand Estimate</u>
Chord, ft	3.48	5.00	7.58
Empty Weight/ Gross Weight	0.542	0.441	0.436
Operating Cost, \$/Ton-NM	0.733	0.865	0.771
Operating Cost, \$/hr	445	525	468
Flyaway Cost, \$ x 10 ⁻⁶	2.128	2.125	1.921

Note: 21 Maintenance manhours per flight hour used for all three cases.

5-2 ARMY SURVEILLANCE AIRCRAFT

A Composite jet-flap stopped rotor aircraft, sized for the typical missions illustrated in Figure 5-1, is illustrated in Figure 5-2. As noted, this aircraft is equipped with a 16-ft. radius, 2-bladed rotor, which is stowed in the upper fuselage for the fixed wing portion of the flight. Two J85-GE-15 engines (less afterburner) are the basic gas generators and are mounted on the aft fuselage. During hover, the exhaust of the engines is directed to the rotor. During transition to forward flight the rotor is unloaded and the exhaust is diverted through the jet nozzles for fixed wing cruise. Characteristics of the vehicle are shown in Table 5-2.

Table 5-2

Surveillance Aircraft Characteristics

Gross Weight	16,700 lb
Length	48 ft
Wing Span	38 ft

Table 5-2 (Concluded)

Surveillance Aircraft Characteristics

Wing Area	278 sq. ft.
Rotor:	
Radius	16 ft
Chord	3.5 ft
Solidity	0.14
Disk loading	20.8 psf
Blade loading	150 psf
$\bar{C}_{L \max}$	1.0
$V_{\max}$ at Sea Level	394 kts
Best Loiter Speed at 9000 lb	190 kts
Maximum Range Speed at SL	200 kts
Transition Speed	100 kts
Wing Loading	60 psf

A weight summary is presented in Table 5-3.

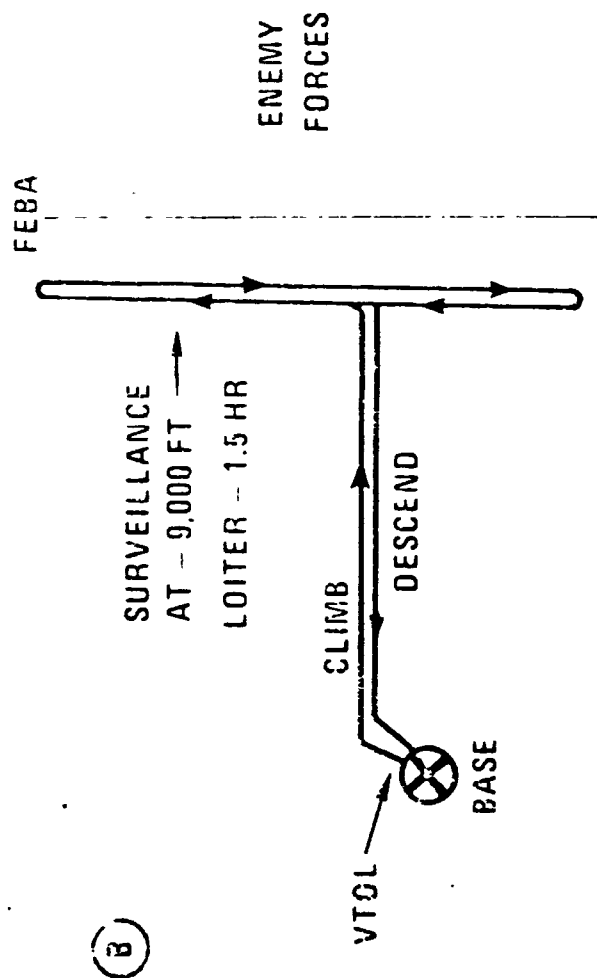
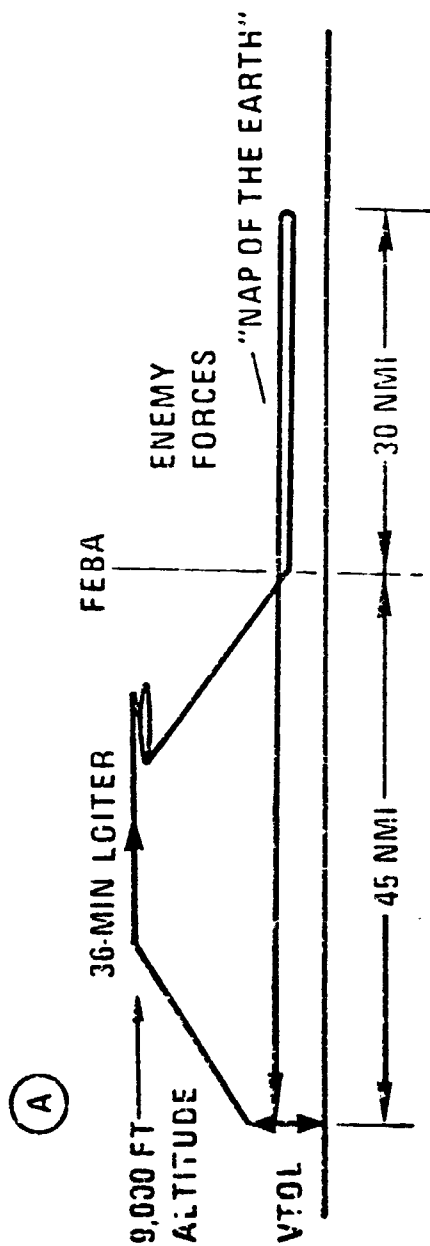


Figure 5-1 Surveillance Mission Profiles

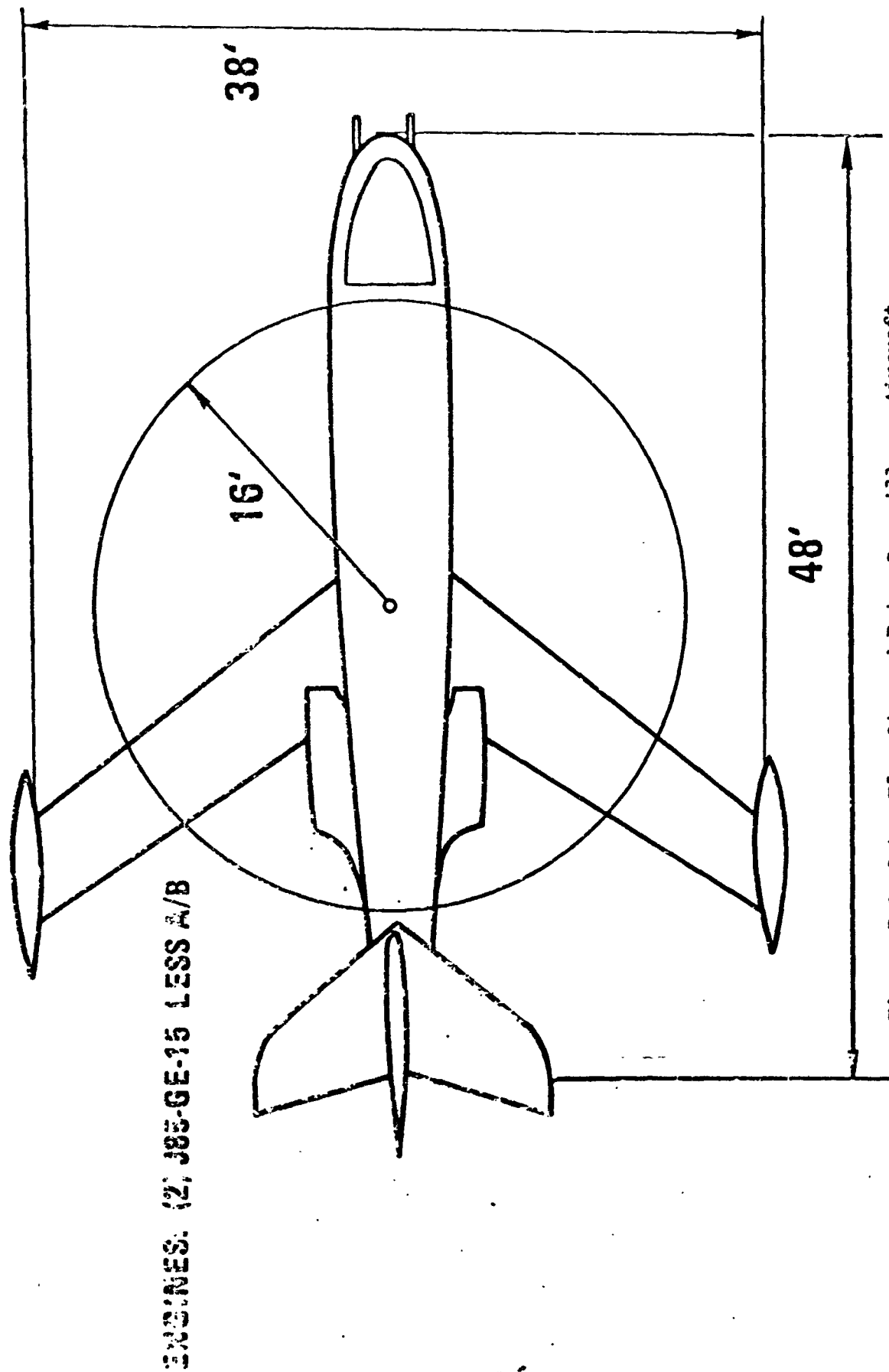


Figure 5-2 Jet - Flap Stopped Rotor Surveillance Aircraft

Table 5-3

Surveillance Aircraft Weight Summary

<u>Structure</u>	
Wing	995
Rotor Group	1782
Horizontal Tail	195
Vertical Tail	78
Fuselage	1433
Main Gear	640
Nose Gear	224
Surface Controls	535
Nacelles	<u>293</u>
Total	6175 lbs
<u>Propulsion</u>	
Engines	784
Air Induction	57
Exhaust	1609
Lubrication System	25
Fuel System	230
Engine Controls	140
Starting System	<u>80</u>
Total	2816 lbs

Table 5-3 (Concluded)

Surveillance Aircraft Weight Summary

<u>Equipment</u>	
Instruments and Navigation	137
Hydraulic	120
Electrical	250
Furnishings	<u>242</u>
Total	749 lbs
Total Empty Weight	9740 lbs
<u>Useful Load</u>	
Crew (2)	400
Oil	20
Fuel - trapped	40
- useable	4000
Payload	<u>2500</u>
Total Useful Load	6960 lbs
Total Take-Off Weight	16,700 lbs

5.3 RESCUE MISSION

For a mission which requires hovering as well as a high speed, long-range capability, such as a rescue mission, the jet-flap rotor, utilized in a stopped rotor composite aircraft offers high potential. A preliminary estimate of the characteristics of such a vehicle has been made for a typical long-range high-speed mission.

5.3.1 Mission (Hot Day Except When Noted Otherwise)

Vertical takeoff at 3,000 feet 95°F

Climb to 30,000 feet

Cruise out at 400 knots at 39,000 feet for a radius of
500 nautical miles

Let down at 7,000 feet

Hover at 100 knots at 7,000 feet for 30 minutes

Hover at 6,000 feet 95°F for 30 minutes, lifting 1,200
pounds after 15 minutes of hover

Return to 20,000 feet

Return to 400 knots at 30,000 feet

5.3.2 Vehicle Description

The configuration of the jet-flap stopped-rotor composite aircraft is illustrated in Figure 5-3. As noted, this aircraft is equipped with a 50-foot diameter two-bladed rotor, which is stowed in the upper fuselage for the fixed wing portion of the flight. The rescue compartment and equipment is located forward of the rotor mast. Two advanced gas generators of 1975 technology are submerged side-by-side in the aft fuselage. Their exhaust may be directed to drive the rotor for rotor-borne flight or

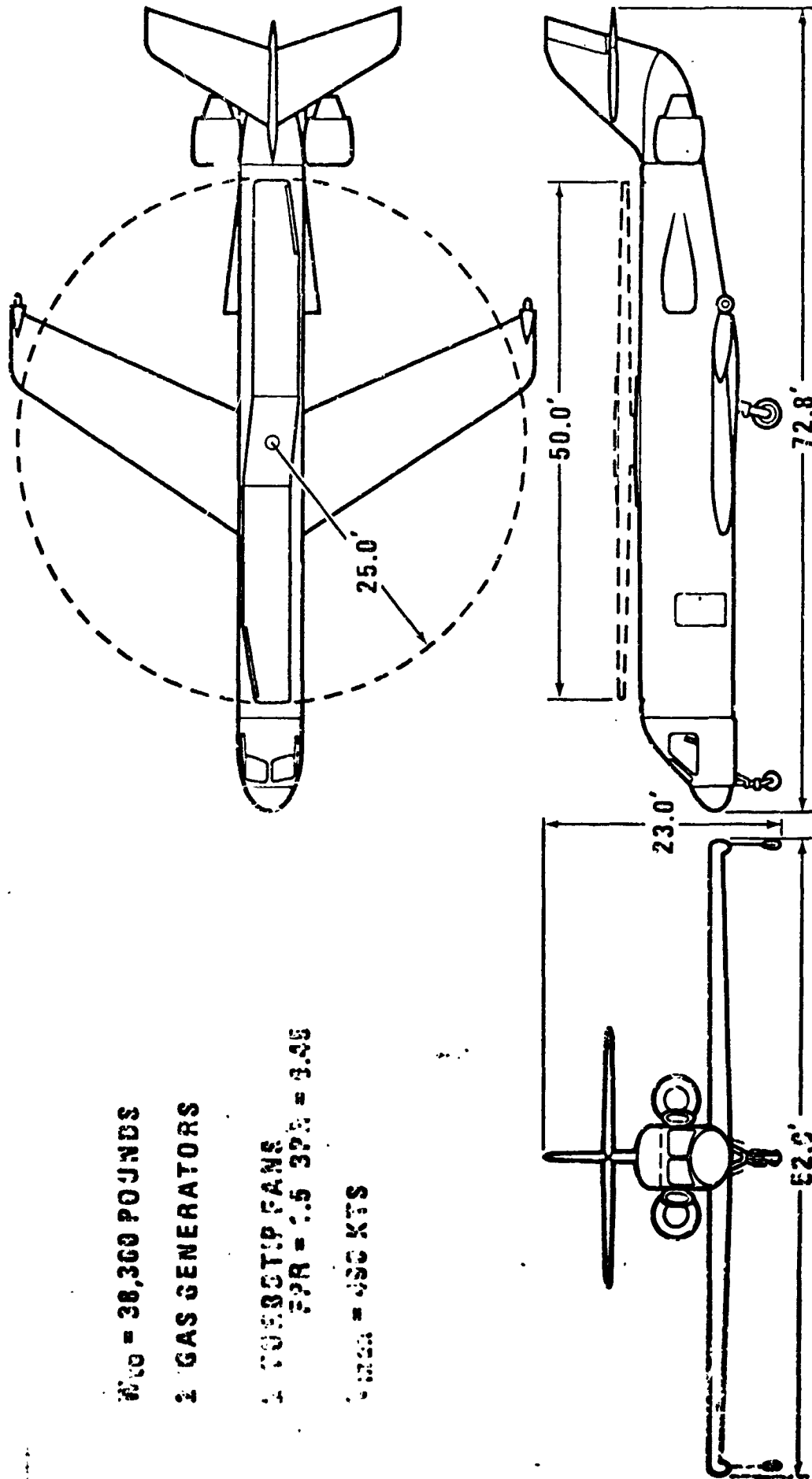
W₀ = 38,300 POUNDS

2 GAS GENERATORS

2 COMBUSTION FANS

FPR = 1.5 325 = 3.45

CRUISE = 450 KTS



PROPRIETARY INFORMATION

Figure 5-3 Jet - Flap Advanced Rescue Airplane

to tip turbofans mounted on the aft fuselage for fixed wing flight and during transition. Characteristics of the aircraft are presented in Table 5-4, and the weight is summarized in Table 5-5. The gross weight of this aircraft is considerably less than any other aircraft concept that can perform the total mission.

5.4 SURVEILLANCE RADAR PLATFORM

Giravions Dorand has been conducting a design study for the French DOD, C.P.E./D.R.M.E. of an unmanned jet-flap vehicle. The design mission is to hover at 16,400 feet (5,000 meters) for 5 hours with a radar payload of 3,670 pounds. The airframe is basically spherical and uses bleed air to rotate the vehicle continuously for radar scanning and for control of its geographical position.

Characteristics of the vehicle are summarized below.

Weight Analysis

Payload	3,670 pounds
Rotor	1,060 pounds
Operating Weight Empty	3,370 pounds
Fuel	4,840 pounds
Gross Weight	11,280 pounds
Maximum pressure ratio	1.88
Gas temperature	650°R
Thrust	1,560 pounds
Fuel Consumption	1,110 pounds/hour
Gas Flow	34.4 pounds/second
CHP	2,160 horsepower

Rotor Characteristics

No. of Blades	2
Diameter	55.8 feet
Profile	NACA64024
Chord	2.95 feet
Solidity	0.0675
Area Ratio	0.76
Blade Duct Mach Number	0.3
Duct Loss Coefficient	$K_T = 0.6$

Engine

SNECMA-TURCOMECA-TUREOFAN M49 "LARZAC"

Bypass ratio 1.4

Installed Engine Weight 660 pounds

Conditions at 16,400 feet

Standard Day:

In this application, with no requirement for pilot or crew, the engine installation and ducting system can be optimized because of the specialized nature of the system. The engine is mounted vertically with exhaust up, requiring only a single 90-degree bend in the ducting. With the gas drive, the empty weight of the vehicle is considerable less than would be required for a shaft-driven system, according to Giravions-Dorand studies.

Table 5-4 Air Rescue Aircraft Characteristics

<u>GROSS WEIGHT WING</u>	38,300 lb
Wing Area	428 Sq. Ft.
Span	49 Ft.
Chord	8.75 Ft
Aspect Ratio	5.6
Thickness Ratio	0.12
Sweep	30°
Wing Loading	
Takeoff	90 PSF
Cruise	80 PSF
Flaps	Single Slotted Fowler
Horizontal Tail Area	111 Sq. Ft.
Vertical Tail Area	62 Sq Ft.
Engine Rated Thrust, SLS	6,190 lb
Transition Speed	130 Kt
Ultimate Load Factor	4
<u>ROTOR</u>	
Diameter	50 Ft.
Solidity	0.115
No. of Blades	2
Chord	4.5 Ft.
Disc Loading	19.5 PSF
Blade Loading	170 PSF
Tip Speed	700 Ft/s
Wing Area/Rotor Area	0.218
Thickness Ratio	0.24

Table 5-5 Air Rescue Aircraft

<u>WEIGHT BREAKDOWN</u>	
Rotor Assembly	3,007
Tail	474
Fuselage	3,952
Wing	2,129
Landing Gear	1,086
Surface Controls	1,160
Engine Nacelles	<u>350</u>
Total Structure	12,184
Engines	1,350
Ducting	2,182
Fuel and Lubricating Systems	1,952
Starting and Controls	140
Fans	<u>1,500</u>
Total Propulsion	7,125
Instruments and Nav	283
Hydraulics and Pneumatic	95
Electrical and Electronics	1,133
Armament	947
Furnishings	313
Air-Conditioning	<u>142</u>
Total Fixed Equipment	3,413
Total Empty Weight	22,722

Table 5-1 (Concluded)

Long Range HLH Characteristics

	<u>Shaft-Driven</u>	<u>LTV Estimate</u>	<u>Dorand Estimate</u>
Chord, ft	3.48	5.00	7.58
Empty Weight/ Gross Weight	0.542	0.441	0.436
Operating Cost, \$/Ton-NM	0.733	0.865	0.771
Operating Cost, \$/hr	445	525	468
Flyaway Cost, \$ x 10 ⁻⁶	2.128	2.125	1.921

Note: 21 Maintenance manhours per flight hour used for all three cases.

5-2 ARMY SURVEILLANCE AIRCRAFT

A Composite jet-flap stopped rotor aircraft, sized for the typical missions illustrated in Figure 5-1, is illustrated in Figure 5-2. As noted, this aircraft is equipped with a 16-ft. radius, 2-bladed rotor, which is stowed in the upper fuselage for the fixed wing portion of the flight. Two J85-GE-15 engines (less afterburner) are the basic gas generators and are mounted on the aft fuselage. During hover, the exhaust of the engines is directed to the rotor. During transition to forward flight the rotor is unloaded and the exhaust is diverted through the jet nozzles for fixed wing cruise. Characteristics of the vehicle are shown in Table 5-2.

Table 5-2

Surveillance Aircraft Characteristics

Gross Weight	16,700 lb
Length	48 ft
Wing Span	38 ft

Table 5-2 (Concluded)

Surveillance Aircraft Characteristics

Wing Area	278 sq. ft.
Rotor:	
Radius	16 ft
Chord	3.5 ft
Solidity	0.14
Disk loading	20.8 psf
Blade loading	150 psf
$\bar{C}_{L \max}$	1.0
$V_{\max}$ at Sea Level	394 kts
Best Loiter Speed at 9000 lb	190 kts
Maximum Range Speed at SL	200 kts
Transition Speed	100 kts
Wing Loading	60 psf

A weight summary is presented in Table 5-3.

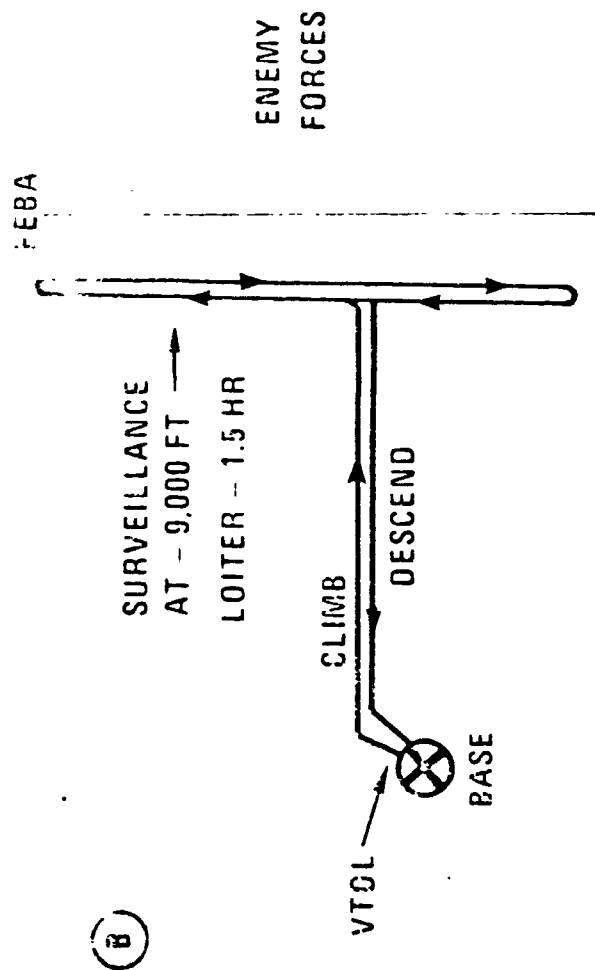
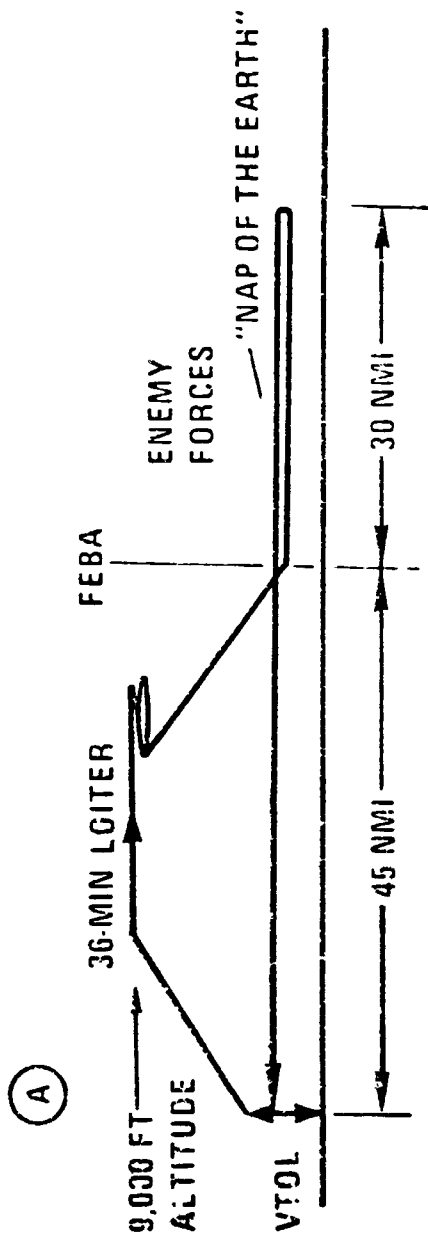


Figure 5-2 Surveillance Mission Profiles

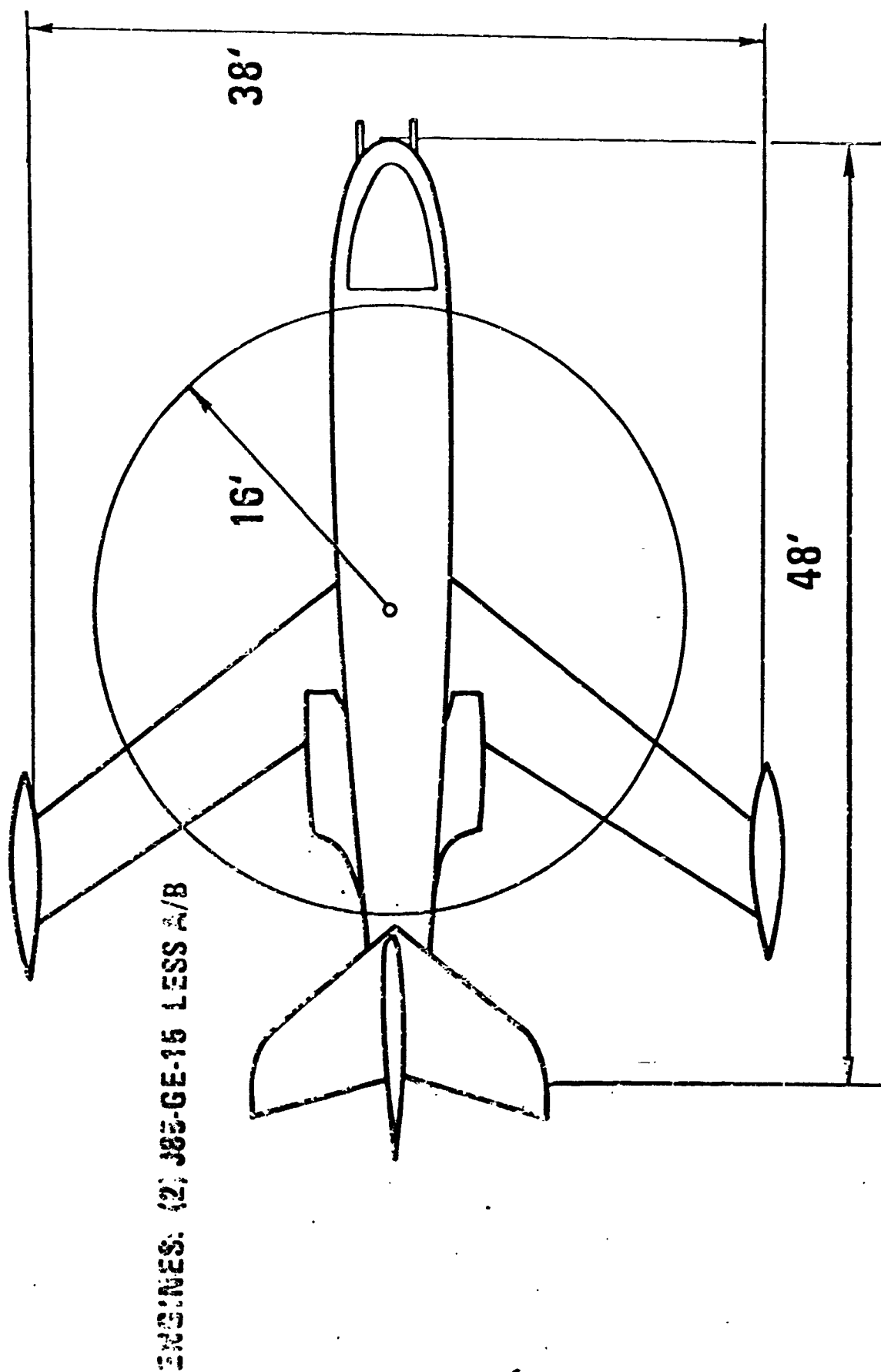


Table 5-3

Surveillance Aircraft Weight Summary

<u>Structure</u>	
Wing	995
Rotor Group	1782
Horizontal Tail	195
Vertical Tail	78
Fuselage	1433
Main Gear	640
Nose Gear	224
Surface Controls	535
Nacelles	<u>207</u>
Total	6175 lbs
<u>Propulsion</u>	
Engines	784
Air Induction	57
Exhaust	1600
Lubrication System	25
Fuel System	230
Engine Controls	40
Starting System	<u>80</u>
Total	2816 lbs

Table 5-3 (Concluded)

Surveillance Aircraft Weight Summary

<u>Equipment</u>	
Instruments and Navigation	137
Hydraulic	120
Electrical	250
Furnishings	<u>242</u>
Total	749 lbs
Total Empty Weight	9740 lbs
<u>Useful Load</u>	
Crew (2)	400
Oil	20
Fuel - trapped	40
- useable	4000
Payload	<u>2500</u>
Total Useful Load	6960 lbs
Total Take-Off Weight	16,700 lbs

5.3 RESCUE MISSION

For a mission which requires hovering as well as a high speed, long-range capability, such as a rescue mission, the jet-flap rotor, utilized in a stopped rotor composite aircraft offers high potential. A preliminary estimate of the characteristics of such a vehicle has been made for a typical long-range high-speed mission.

5.3.1 Mission (Hot Day Except When Noted Otherwise)

Vertical takeoff at 3,000 feet 95°F

Climb to 30,000 feet

Cruise out at 400 knots at 39,000 feet for a radius of
500 nautical miles

Let down at 7,000 feet

Hover at 100 knots at 7,000 feet for 30 minutes

Hover at 6,000 feet 95°F for 30 minutes, lifting 1,200
pounds after 15 minutes of hover

Return to 20,000 feet

Return to 400 knots at 30,000 feet

5.3.2 Vehicle Description

The configuration of the jet-flap stopped-rotor composite aircraft is illustrated in Figure 5-3. As noted, this aircraft is equipped with a 50-foot diameter two-bladed rotor, which is stowed in the upper fuselage for the fixed wing portion of the flight. The rescue compartment and equipment is located forward of the rotor mast. Two advanced gas generators of 1975 technology are submerged side-by-side in the aft fuselage. Their exhaust may be directed to drive the rotor for rotor-borne flight or

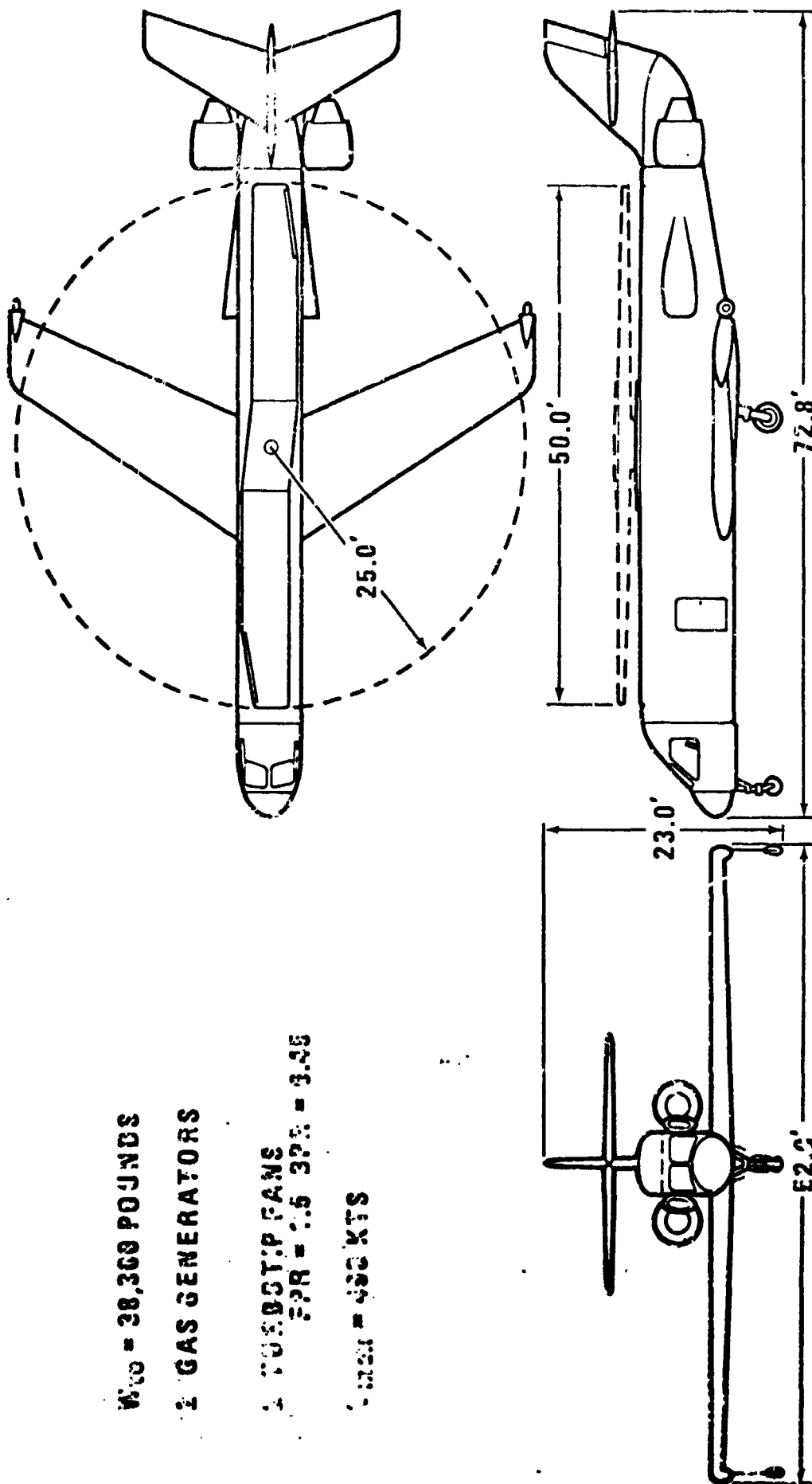
W₀ = 38,360 POUNDS

2 GAS GENERATORS

2 TURBOJET PUMPS

FPR = 1.5 320 = 3.35

CRUISE = 430 KTS



PROPRIETARY INFORMATION

Figure 5-3 Jet - Flap Advanced Rescue Airplane

to tip turbofans mounted on the aft fuselage for fixed wing flight and during transition. Characteristics of the aircraft are presented in Table 5-4, and the weight is summarized in Table 5-5. The gross weight of this aircraft is considerably less than any other aircraft concept that can perform the total mission.

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Giravions Dorand has been conducting a design study for the French DOD, C.P.E./D.R.M.E. of an unmanned jet-flap vehicle. The design mission is to hover at 16,400 feet (5,000 meters) for 5 hours with a radar payload of 3,670 pounds. The airframe is basically spherical and uses bleed air to rotate the vehicle continuously for radar scanning and for control of its geographical position.

Characteristics of the vehicle are summarized below.

Weight Analysis

Payload	3,670 pounds
Rotor	1,060 pounds
Operating Weight Empty	3,370 pounds
Fuel	4,840 pounds
Gross Weight	11,280 pounds
Maximum pressure ratio	1.88
Gas temperature	650°F
Thrust	1,560 pounds
Fuel Consumption	1,110 pounds/hour
Gas Flow	34.4 pounds/second
CHP	2,160 horsepower

Rotor Characteristics

No. of Blades	2
Diameter	55.8 feet
Profile	NACA64024
Chord	2.95 feet
Solidity	0.0675
Area Ratio	0.76
Blade Duct Mach Number	0.3
Duct Loss Coefficient	$K_T = 0.6$

Engine

SNECMA-TURCOMECA-TUREOFAN M49 "LARZAC"	
Bypass ratio	1.4
Installed Engine Weight	660 pounds
Conditions at 16,400 feet	
Standard Day:	

In this application, with no requirement for pilot or crew, the engine installation and ducting system can be optimized because of the specialized nature of the system. The engine is mounted vertically with exhaust up, requiring only a single 90-degree bend in the ducting. With the gas drive, the empty weight of the vehicle is considerable less than would be required for a shaft-driven system, according to Giravions-Dorand studies.

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Thickness Ratio	0.12
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Flaps	Single Slotted Fowler
Horizontal Tail Area	111 Sq. Ft.
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Engine Rated Thrust, SLS	6,190 lb
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Diameter	30 Ft.
Solidity	0.115
No. of Blades	2
Chord	4.5 Ft.
Disc Loading	19.5 PSF
Blade Loading	170 PSF
Tip Speed	700 FPS
Wing Area/Rotor Area	0.218
Thickness Ratio	0.4

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Rotor Assembly	3,007
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Fuselage	3,952
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Surface Controls	1,160
Engine Nacelles	<u>350</u>
Total Structure	12,184
Engines	1,350
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Fuel and Lubricating Systems	1,952
Starting and Controls	140
Fans	<u>1,500</u>
Total Propulsion	7,125
Instruments and Nav	283
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Electrical and Electronics	1,133
Armament	957
Furnishings	313
Air-Conditioning	<u>142</u>
Total Fixed Equipment	3,413
Total Empty Weight	22,722

Table 5-5 Air Rescue Aircraft (Concluded)

<u>Weight Breakdown</u>	
Crew	860
Fuel	14,323
Oil	105
Ammo (6000 Rd)	220
Mini Guns (2)	<u>70</u>
Total Useful Load	15,578
Gross Takeoff Weight	38,300

6.0 COST EFFECTIVENESS ANALYSIS

6.1 BASIC COST EFFECTIVENESS CRITERIA

In studies of this type, evaluation criterion must of necessity be selected which can be fitted to the design and performance data to be developed. Thus, the sophistication of the cost effectiveness model is dependent on the depth and detail to which the design is developed. Initially, designs can be optimized parametrically using weight empty as a cost parameter and gross weight as the effectiveness parameter. As the design progresses, effectiveness parameters such as ton miles per hour, ton miles per pound of fuel, tons per hour, etc. are used. At the same time complexity factors may be applied to various areas of cost for more rational cost criterion. However, experience has shown that the cost of RDT&E and hardware on new development programs in which the state-of-the-art is extended cannot be estimated with very much confidence, even when considerable design detail is available.

The cost effectiveness criteria selected for comparing the various HLM and HSH configurations are:

- . Operating Cost per Ton Nautical Mile
- . Operating Cost per Flight Hour

Operating cost is used in lieu of total life cycle cost because, 1) it is more directly related to aircraft performance and capability, 2) it is much less sensitive to variations in cost of RDT&E and hardware for which estimates can not be made with very much confidence at this stage and 3) it is a measure with which the contractor is familiar and experienced.

The equation for operating cost was developed originally for preliminary evaluations and parametric analyses on the Light Intratheater Transport (LIT) VTOL and STOL studies. It is based on specified Air Force operational and cost factors and has been reduced to the following basic equation.

$$C_o = [164 + 42.11 R_c P'_c + .025 UM + 33.1 H_D C_f (10^{-6}) + .014425 H_D F_H] \frac{1}{H_D}$$

where C_o = Operating Cost - \$ per flight hour - (1967 dollars)

R_c = Number of flight crews per A/C = 1.5

P'_c = Number of men per flight crew = 2.0

U = Annual utilization in hours per A/C = 1000

M = Maintenance manhours per flight hour

H_D = Flight hours per day per A/C = $1000/365 = 2.74$

C_f = Airplane unit cost in dollars

F_H = Fuel used per A/C per flight hour in pounds

Since the above equation requires aircraft unit cost as an input, an empirical equation for estimating flyaway cost is provided. This equation was derived in early 1968 by a regression analysis of ten data points; two on conventional aircraft and eight on V/STOL designs which were previously costed in detail. When the resulting equation was applied to a current production helicopter design, the results were close to available values. Although the specific values obtained from the equation may be open to question, it does provide a standard basis for evaluating differential cost effects of basic design and performance variations.

The equation is:

$$C_f = 547(M_{CR})^{.01725} (T/H)^{.2314} \left(\frac{W}{W_0}\right)^{.4884} \left(\frac{N}{.001W}\right)^{.61} \left(\frac{.001W}{S}\right)^{.6397} \left(\frac{WE}{1000}\right)^{.3881}$$

where C_f' = Airplane cost per pound of empty weight including engines and electronics @ 200 units in 1967 dollars.

M_{CR} = Cruise Mach number

T/W_t = S.L. std. day thrust to weight ratio

W_a = $WE - (W_e + W_{el})$

WE = Weight empty

W_e = Weight of engines

W_{el} = Weight of electronics

W_t = Gross takeoff weight

N_Z = Ultimate load factor = 3.75

N_e = Number of engines

S = Equivalent wing area = $2RCB$

R = Rotor radius - ft.

C = Rotor average chord - ft.

B = Number of blades

$$C_f = (C_f')(WE)(3.682 \times N_{A/C}^{-.246})$$

where C_f = Cumulative average total airplane cost per airplane in 1967 dollars

$N_{A/C}$ = Number of aircraft produced = 100

In the initial cost effectiveness evaluation, an estimate of 10 MCH/FH was used for both the jet flap and shaft driven HSH. However, as more detailed information became available, preliminary estimates of the maintenance manhours per flight hour were made for the LTV hot cycle HSH point design and the shaft driven HSH point design. Results indicated 11.6 maintenance manhours per flight hour were required for the jet-flap helicopter and 13.0 for the shaft-driven vehicles. With a cold cycle

jet-flap it is estimated that the two would be approximately the same due to the additional machinery associated with the cold cycle. A preliminary estimate indicated that 18 maintenance manhours per flight hour would be required for the shaft driven and the cold cycle Heavy Lift helicopters. A breakdown of the High Speed estimated maintenance is summarized below.

<u>Organizational Maintenance (MH/FH)</u>	<u>Jet-Flap</u>	<u>Shaft</u>
Scheduled Maintenance	2.50	3.00
Unscheduled Maintenance		
Aircraft Basic	(2.00)	(2.50)
Engines	(2.50)	(3.00)
Utilities	(0.35)	(0.25)
Instruments	(0.25)	(0.25)
Communications	(0.50)	(0.50)
Navigations	(0.45)	(0.45)
Miscellaneous	<u>(0.05)</u>	<u>(0.05)</u>
Total Unscheduled	<u>6.10</u>	<u>7.00</u>
Total Organizational	8.60	10.00
<u>Intermediate Maintenance</u>	<u>3.00</u>	<u>3.00</u>
Total Maintenance	11.60	13.00

6.2 COST EFFECTIVENESS COMPARISONS

Based on the foregoing criteria and methodology. Figure 6-1 compares the cost effectiveness in dollars per flight hour and dollars per ton nautical mile of various HSH jet flap concepts with the shaft driven HSH and also considers the effects of cruise speed variation on the shaft

HIGH SPEED HELICOPTER OPERATING COST EFFECTIVENESS VS CRUISE SPEED

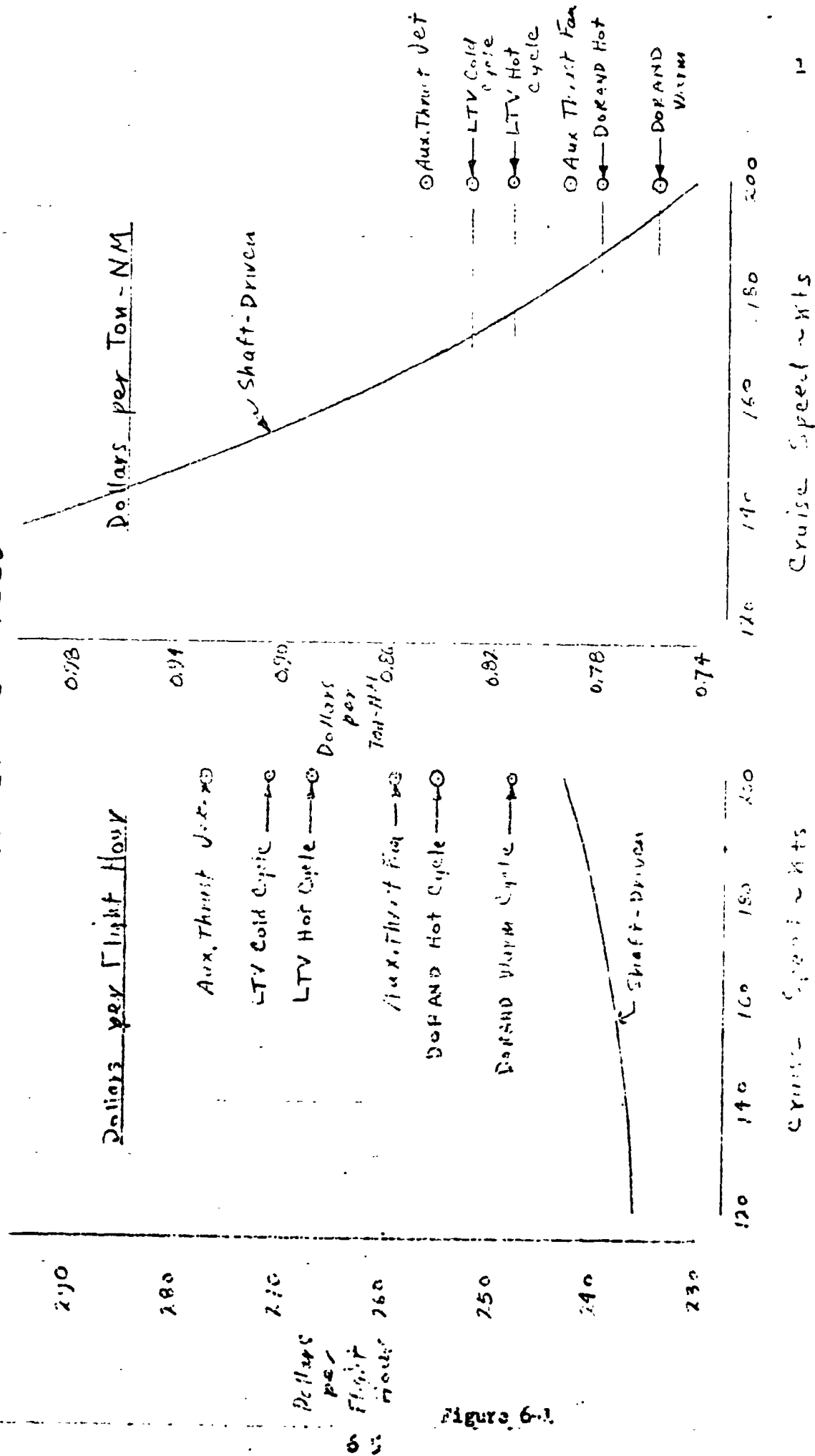


Figure 6-1

driven helicopter. Because with the mechanical system, alleviation of rotor cyclic loading (vertical vibration) is not readily achieved, as is the case with the jet flap, practical considerations may limit the continuous cruise speed of the shaft-driven vehicle. The operating cost varies drastically with speed as indicated in Figure 6-1. If the jet flap at 200 kts is compared with the shaft-driven helicopter at lower speeds, an operating cost advantage in dollars per ton NM for the jet flap can be realized. Cost per flight hour remains higher for the jet flap. In this figure estimates of maintenance manhours per flight hour were taken as 10.0 for both the jet-flap and shaft-driven vehicle.

The sensitivities of key jet flap rotor parameters to operating costs in dollars per ton NM are presented in Figure 6-2. This figure shows the effects of variations in thrust recovery, duct loss, parasite drag and maintenance manhours per flight hour. It is evident that when compared at 200 knot cruise speed, the shaft-driven helicopter has lower operating cost than the jet-flap helicopter for reasonably expected values of the parameters.

Cost effectiveness figures for several HSH and HLH are tabulated in Table 4-2 and 4-3.

6.3 ALTERNATE COST EFFECTIVENESS CRITERIA

During the latter phase of the Jet Flap Rotor Application Study it was decided to compare the Jet Flap and Shaft Driven helicopters on the basis of total system life cycle cost using a ten year system life. However, because of the uncertainty associated with estimating RDT&E and hardware, the aircraft were costed independently by two different groups. One group used the methodology of Paragraph 6.1 to

"REPRODUCIBILITY OF THE ORIGINAL 'A-H-K' POOR."

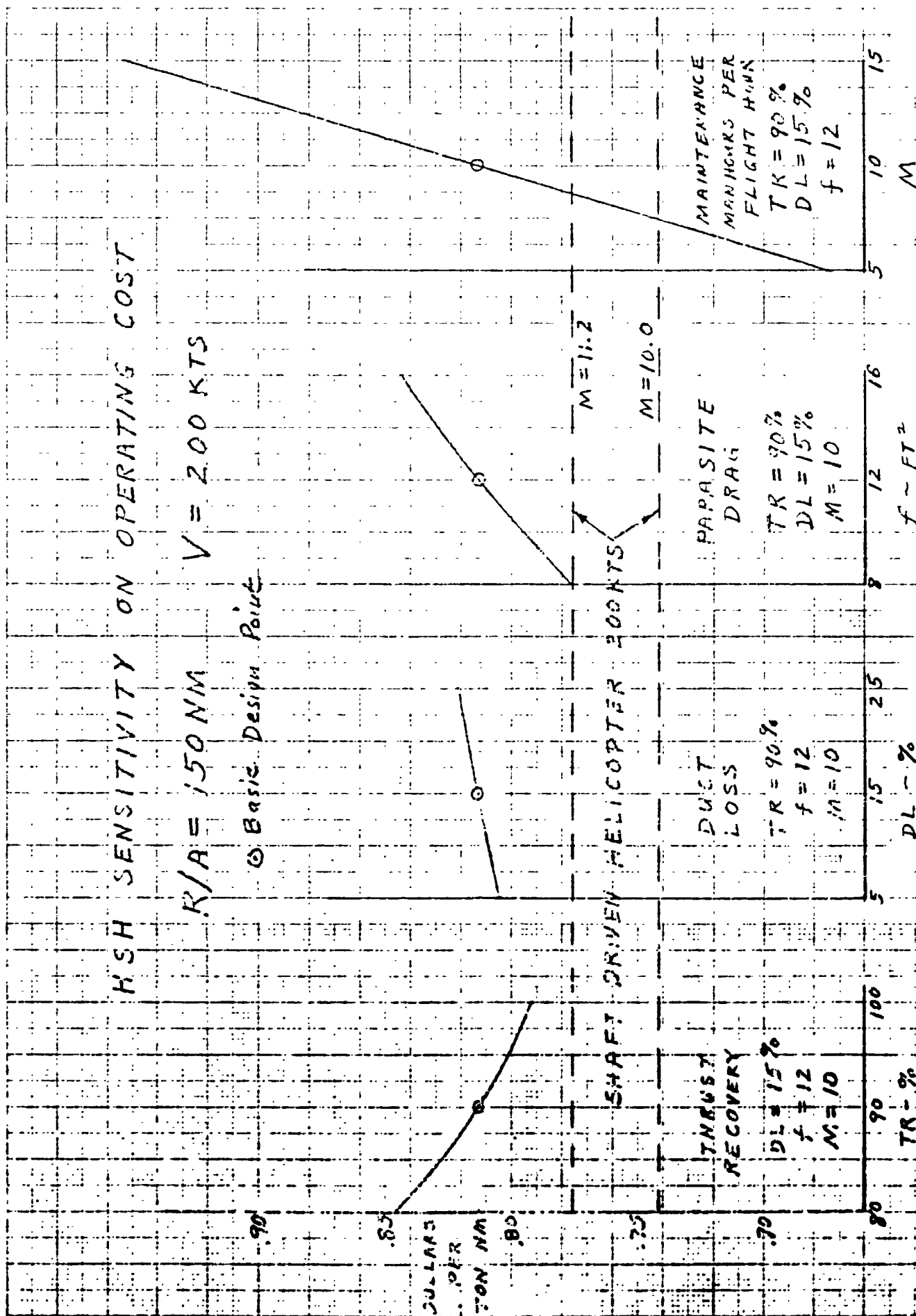


Figure 6-2.

determine operating cost and flyaway cost, plus the following equations to determine total acquisition and RDT&E. Again Air Force factors were used from the LIT study.

Acquisition Cost

$$C_{aq} = 1.30 C_f N_{A/C} + (112,000 + 11,100 R_C P_C' + 16.35 UM) N_{UE}$$

C_{aq} = Total system acquisition cost - dollars

C_f = Flyaway cost per A/C

$N_{A/C}$ = Number of aircraft in initial procurement

N_{UE} = Number of U.E. aircraft in initial procurement

R_C = Flight crew ratio (no. of crews per A/C)

P_C' = Number of personnel in a flight crew

U = Annual flight hours per aircraft

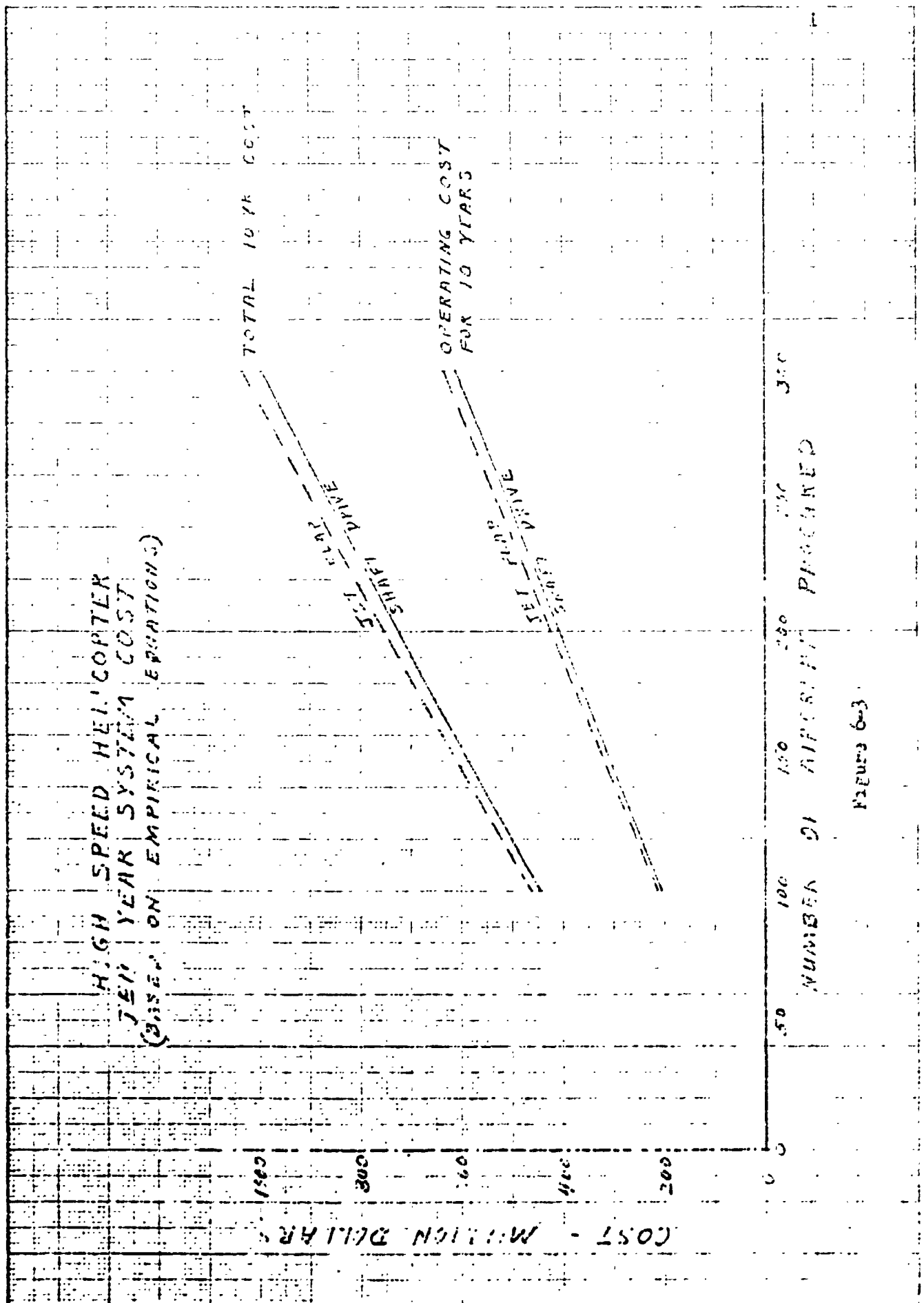
M = Maintenance manhours per flight hour (w/o depot)

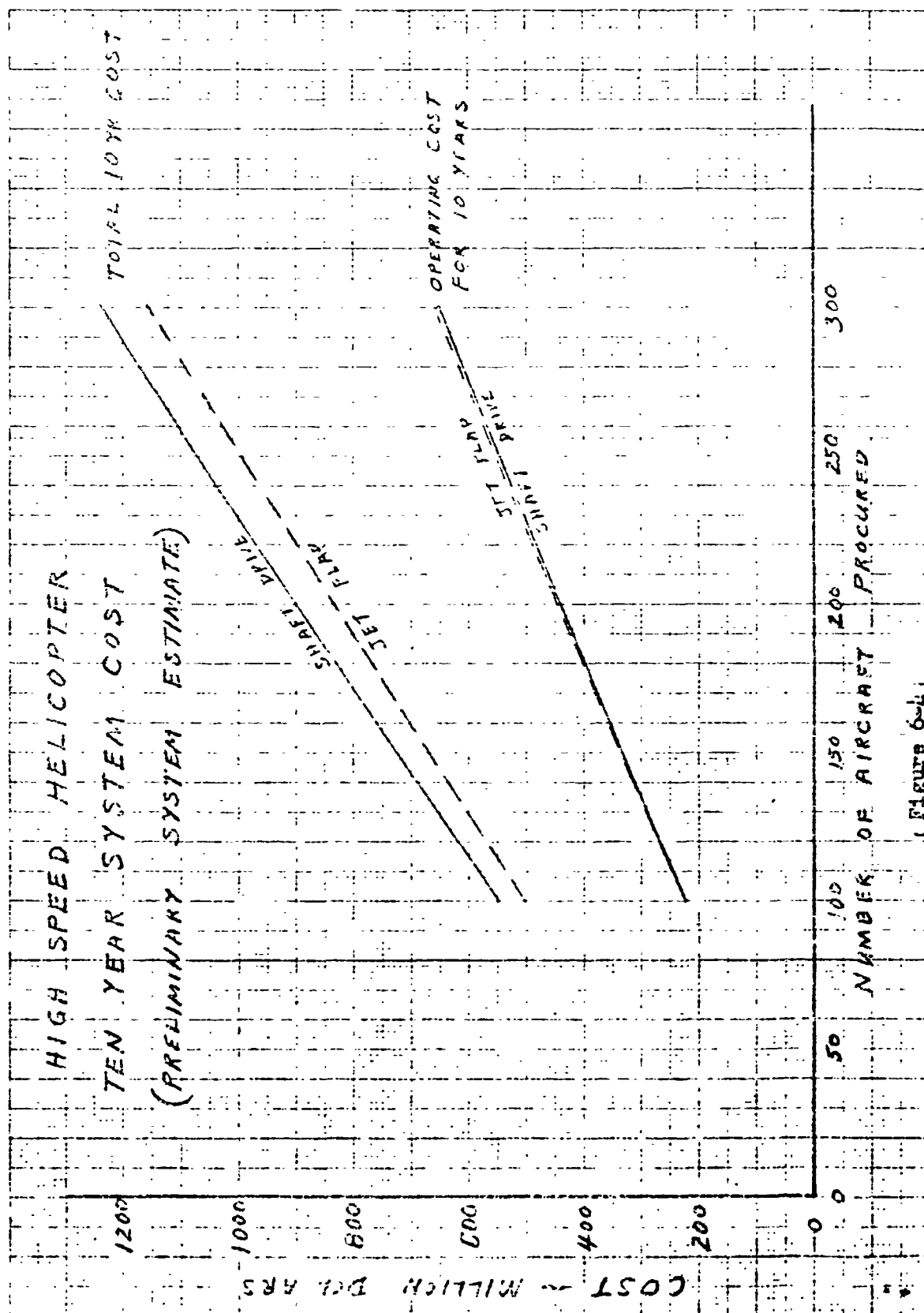
$$C_{RDT\&E} = 95 \times 10^6 + .3675 (200 C_f)$$

C_f = Cum. average flyaway price at 200

The other group used recent historical A-7 data such as manhours per pound to the maximum extent possible. In addition, general estimating reports developed by Rand Corporation were used to estimate propulsion system costs, and for areas with limited experience such as systems engineering or data, cost methodologies from LIT studies and the C-142 cost proposal were used.

Total RDT&E, acquisition, and operating cost for ten years estimated by the independent methods described above were combined into Total System Cost and the results are compared in Figures 6-3 and 6-4. In comparing Figures 6-3 and 6-4, both methodologies show very little to





cost between the Jet Flap and the Shaft Drive from the standpoint of operating cost over a 10 year period. However, if total ten year cost is the criterion, Figure 6-3 (based on empirical equations) shows the Jet Flap about four (4) percent higher cost than the Shaft Drive while Figure 6-4 shows the Jet Flap about seven (7) percent lower cost than the shaft drive. Comparing absolute numbers for a procurement of 100 aircraft shows a difference in total system cost between the two methodologies of about 12% for the jet flap and 24% for the shaft drive while operating costs show a variation of only 3% for the jet flap and 7% for the shaft drive. This illustrates one reason for selecting operating cost as the basic evaluation criterion.

No clear cut advantage for either system based on cost effectiveness can be predicted based on these studies. Additional design and operating data are needed to achieve a more confident cost effectiveness comparison.

7.0 CONCLUSIONS

7.1 HIGH-SPEED HELICOPTER COMPARISON

Assuming optimistic values for thrust recovery factor and optimistic aerodynamic and propulsion system designs, the jet-flap high-speed helicopter can become competitive with the shaft-driven helicopter of comparable technology in terms of gross weight and possess a significantly lower empty weight. From a cost-effectiveness standpoint at the same speed of 200 kts, no clear-cut advantage for the jet-flap is evident, as its operating cost is higher due to higher fuel consumption which tends to offset potential flyaway cost savings. The flying qualities of the jet-flap helicopters at high speed are significantly smoother than the conventional shaft-driven vehicle. This feature could contribute to lower maintenance requirements, more favorable fatigue environment for the structure and pilot, and longer-lived components, all of which could further decrease the total life cycle cost of the jet-flap relative to the shaft-driven helicopter. In the present study, quantitative accountability for these types of features in the cost was not possible. However, if the shaft-driven vehicle were limited to cruise speeds below about 160 knots the jet-flap operating cost is less.

Of the various compound designs examined, the tip-turbine direct thrust and the split-flow tip nozzle concepts showed a superiority over the pure jet-flap helicopter in weight and operating cost. Other compound concepts examined, including the direct thrust jet nozzle, and direct thrust turboprop were heavier than the pure jet-flap helicopter. None of the pneumatic concepts achieved an operating cost advantage over the shaft driven concept under the ground rules of this study.

7.2 HIGH SPEED POTENTIAL

Some results of the analysis of the high speed potential of the jet-flap rotor are summarized below:

- . Maximum speed capability of jet-flap rotor configured helicopters is limited by power requirements rather than by blade aerodynamics as is the case for a shaft-driven helicopter.
- . Forward speeds of over 250 kts are aerodynamically possible with a jet-flap rotor.
- . Power considerations based on 6000' 95° sizing will generally limit speeds to under 250 kts.
- . The two-per-rev vertical forces characteristic of 2-bladed rotors can be alleviated in a shaft-driven helicopter by superimposing blowing. Penalty due to mechanical complexity was not evaluated in this study.

7.3 PRESSURE RATIO COMPARISON

In all cases studied, (high-speed, heavy-lift, hot-warm cycle, cold cycle) minimum gross weight occurred at pressure ratios of around four. Most prominent factors contributing to this result in offsetting the favorable propulsion system efficiencies at low pressure ratios are:

- . Duct area required which leads to high solidity and high blade and duct weights at the lower pressure ratios.
- . Gas horsepower-to-engine weight ratio which increases with increasing pressure ratio.

7.4

CYCLE COMPARISON

Because of its simplicity, favorable gas-horsepower-to-weight ratio and its compatibility with higher pressure ratio, the hot cycle yields the lightest gross weight vehicle for a given mission, being approximately 8 percent lighter than the cold cycle at the same pressure ratio. However, the temperature associated with the late 1970's gas generators used in this study are of the order of 1600°F. From a materials point of view, this is a high risk area. In the design of an actual vehicle, the weight penalty for operating at lower pressure ratios and temperature should be evaluated.

7.5

THRUST RECOVERY FACTOR

The achievable value of thrust recovery with a rotating rectangular nozzle is not well known. It is a relatively sensitive parameter in the high-speed case. The sensitivity study indicated that, for a nozzle velocity coefficient of 0.96, average jet-flap aerodynamic thrust recovery values of the order of 0.94 are required to make the jet-flap HSH competitive with the shaft-driven HSH. Improved methods for estimating thrust recovery are needed.

7.6

DUCTING PRESSURE LOSS

For a reasonable compact conventional propulsion system installation (i.e. engines horizontal fore and aft, with two bends of approximately 90° each), the minimum pressure loss $\frac{\Delta P}{P}$ from gas generator exhaust to jet-flap nozzle, is expected to be approximately 0.08 to 0.11 at a pressure ratio of 2 and 0.11 to 0.15 at a pressure ratio of four. The sensitivity of gross weight to pressure loss is approximately one-half percent per percent pressure drop.

7.7 OTHER SENSITIVITIES

Relatively low sensitivity of gross weight to the following parameters was indicated:

- . Area ratio (duct area/total blade profile area) -
 $2\% \frac{\Delta W}{W}$ per 0.10 area ratio.
- . Tip Speed - Negligible change in gross weight of the Heavy Lift Helicopter over the range of 640 to 800 fps tip speed.
- . Direct thrust at high speed - gross weight increase of 14 percent for turbojet cruise propulsion at 200 kts to overcome parasite drag; 3 percent decrease in gross weight for turbofan cruise.
- . Tip nozzle power split - A decrease in gross weight of percent is achievable if 50 percent of the flow is directed through tip nozzles. Whether this would leave adequate flow for control is not known.

7.8 PARASITE DRAG

Gross weight sensitivity to parasite drag area (fuselage, empennage, hub, interference) is approximately 3 percent per square foot drag area. Thus care should be taken in minimizing the drag by aerodynamically "cleaning up" the hub area. However, based on analytical and experimental data, confidence that the minimum achievable equivalent drag area for the jet flap helicopter of this study is 12 ± 1 sq. ft. is high.

7.9 HEAVY-LIFT HELICOPTER COMPARISON

For the very short range mission (10NM radius), both the hot and cold cycle, the 3-bladed jet-flap helicopters were lighter (hot cycle 11.5 percent, cold cycle 4.8 percent) than the shaft-driven helicopter. As in the HSH case, operating cost per ton NM of the jet-flap was slightly higher. An additional improvement in weight of some 6 percent of the jet-flap over the shaft-driven HSH would be realized if the number of blades were reduced to two.

7.10 VALIDITY OF COMPARISON CRITERIA

As indicated in Paragraph 6.1, in a study of this type where the emphasis is on parametric performance rather than detailed design, validity of cost as a comparison criterion must be questioned and placed in proper perspective. Operating cost appears to be a more valid criterion than total life cycle or hardware cost primarily because it is directly related to performance and thus is less sensitive to variations in RDT&E and hardware costs, estimates of which cannot be made with much confidence at this stage. Probably more valid criteria for this comparison are weight and other characteristics and these are also presented. While operating costs per hour, operating cost per ton-NM, and flyaway cost are presented for various cases, it is cautioned that these should only be considered as indicators rather than hard final criteria. The weight and other characteristics should be considered in comparing the merits of the concepts.

7.11 OTHER APPLICATIONS

While only limited studies were conducted on applications other than the two basic applications, results indicate that best

utilization of the high $\bar{C}_L$ capability of the jet-flap appears to be where high disc loadings are required. This is especially true in the stoppable rotor case where rotor size is important. Of course, from a power requirement point-of-view, the tendency is to go to lower disc loadings but in that case, the high $\bar{C}_L$ capability of the jet-flap rotor is not exercised.

The pneumatically-driven rotor, equipped with a jet-flap or similar control device, appears to offer promise in a stopped and stowed rotor application. It's stiffness, continuous control capability, compactness, and light weight provide features which tend to alleviate some of the problems that have been encountered with conventional rotor stopping and stowing. Thus, further investigation of this application of the jet-flap is recommended.