UNCLASSIFIED

AD NUMBER	
AD141370	
CLASSIFICATION CHANGES	
TO:	**UNCLASSIFIED**
FROM:	**CONFIDENTIAL**
LIMITATION CHANGES	
TO: **Approved for public release; distribution is unlimited. Document partially illegible.**	
FROM: **Distribution authorized to U.S. Gov't. agencies and their contractors; Administrative/Operational Use; MAR 1956. Other requests shall be referred to Office of Naval Research, 875 North Randolph Street, Arlington, VA 22203-1995. Document partially illegible.**	
AUTHORITY	
31 Mar 1968, DoDD 5200.10; ONR ltr, 28 Oct 1977	

THIS PAGE IS UNCLASSIFIED

[redacted]

AD 141370

Armed Services Technical Information Agency

Reproduced by

DOCUMENT SERVICE CENTER

KNOTT BUILDING, DAYTON, 2, OHIO

FOR

MICRO-CARD

CONTROL ONLY

[redacted] OF 3

[redacted]

CONFIDENTIAL

Fil 461

FC

AD No. 141320

ASTIA FILE COPY

VTOL TRANSPORT AIRCRAFT

Comparative Study

INTERIM SUMMARY REPORT
VERTOL REPORT NO. R-75

SEP 9 1957

VERTOL Aircraft Corporation

formerly - Piasecki Helicopter Corporation

57AA 42109

CONFIDENTIAL

8-277

CONFIDENTIAL

Comparative Study of Various Types of VTOL Transport Aircraft

INTERIM SUMMARY REPORT R-75

Vertol Aircraft Corporation Morton, Pennsylvania

ONR ARMY

A Combined

Research and Development Program

Contract NONR 1681(00)

[illegible]

PREPARED BY
RESEARCH AND DEVELOPEMENT

Project Engineer J. MALLEN

Supervised by W. Z. Stepniewski
W. Z. STEPNIEWSKI Chief R & D

Approved by L. L. Douglas
L. L. DOUGLAS Vice Pres. - Engineeing

57AA 42109

Copy No. ________ CONFIDENTIAL Mar. 12, 195[illegible]

TABLE OF CONTENTS

I. SUMMARY

In May 1955, Vertol Aircraft Corporation received Contract Nonr 1681(00) from the Office of Naval Research to undertake a broad research study of vertical take-off and landing aircraft suitable for military transport missions in the period 1960 to 1965.

This report, based on a parametric study of the problem is submitted as partial fulfillment of the requirements of the subject contract. It summarises the initial efforts in establishing the relative competitive position of the many configurations conceived for VTOL transport applications.

A. Objectives

The objectives of this phase of the research study are twofold:

1. To compile and consolidate technical data for various VTOL design concepts with particular emphasis on trends of component weights and powerplant data. In keeping with the intent of the study, the trend data will be extrapolated to reflect 1962 state of art.

2. Using the trend data, to evaluate the relative position of the various VTOL design concepts on a technical basis in order to determine the configurations most suitable for the military transport mission. Take-off gross weight was used as the criterion for comparing the VTOL aircraft capable of accomplishing the following specified mission:

a. Payload	8000 lb. out - 4000 lb. back
b. Take-off	Vertical
c. Cabin Size	8' x 9' x *
d. Cargo	35 Infantry troops or equivalent vehicles
e. T.O. Conditions	Pressure altitude 6000 ft. at 95°F
f. Runway Surface	Friction coefficient μ = .2; UCI = 15 **
g. Cruise Speed	300 MPH
h. Flight Profile	20% of radius adjacent to target at S. L.

I. Landing	Vertical
J. Radius of Action	425 Statute miles

*As required to accommodate 35 troops.
**Applicable to the case of running take-off at overload gross weight.

A map of Europe and Asia has been prepared, Figure 1-1, to illustrate the radius of action capabilities of this assault transport. The shaded areas indicate possible areas of application assuming the operation originates from outside the Soviet Union and its satellites.

B. Assumptions

1. Mission Deviations

Several deviations from the specified mission were made in order to evaluate the numerous VTOL design concepts as quickly as possible:

a. Payload - 8000 pounds outbound and inbound

b. Cruise at sea level

c. Cruise at 80% of rated military power

These deviations were made to simplify the calculations and do not effect trends but merely result in conservative (heavy) estimates for take-off gross weight.

2. Engine-out Safety Provisions

Interconnecting shafts have been provided for all applicable design concepts in order to satisfy the requirement that the aircraft remain controllable with one engine inoperative and be able to make a "controlled crash" landing. In addition, weight provisions and power requirements have been estimated to assure positive control throughout all regimes of flight with particular emphasis on the take-off and landing conditions.

3. Time in Hovering

Time spent in hovering is an extremely important factor and has an appreciable effect in determining not only the optimum design parameters for certain design concepts but also effects the comparative competitive position of the various VTOL configurations. After discussing this item with operational personnel, it was determined that an absolute minimum of five (5) minutes of hovering would be required to perform the specified mission.

In addition, a two (2) minute warm-up was assumed. Therefore a total time of seven (7) minutes was used in calculating fuel requirements in hovering.

Since hovering time is an important aspect of VTOL aircraft, the study will be extended in the near future to include the effect of varying the time spent in the hovering regime of flight. The length of hovering time may influence the competitive position of the various VTOL configurations.

4. General

Since the study is for transport application, only those concepts that retain basically a horizontal attitude of the fuselage in all regimes of flight were investigated.

C. Scope

In order to investigate and categorize the many VTOL design concepts, it was decided to consider cruise speed as a variable. With cruise speed as a variable, the entire spectrum of VTOL aircraft, from helicopters to direct-lift turbojet aircraft, can be evaluated. In addition to determining the competitive position of each VTOL configuration, the results of this type of analysis can be used directly to assess the relative effect of cruise speed with regard to the size of any given aircraft.

The VTOL configurations studied may be divided into two distinct categories:

a. Rotary wing concepts

b. Fixed wing concepts

Three basic configurations were investigated under item (a):

1. Conventional helicopter with and without boundary layer control

2. Compound helicopters

3. Retractoplanes

For each configuration several combinations of available powerplants were assumed.

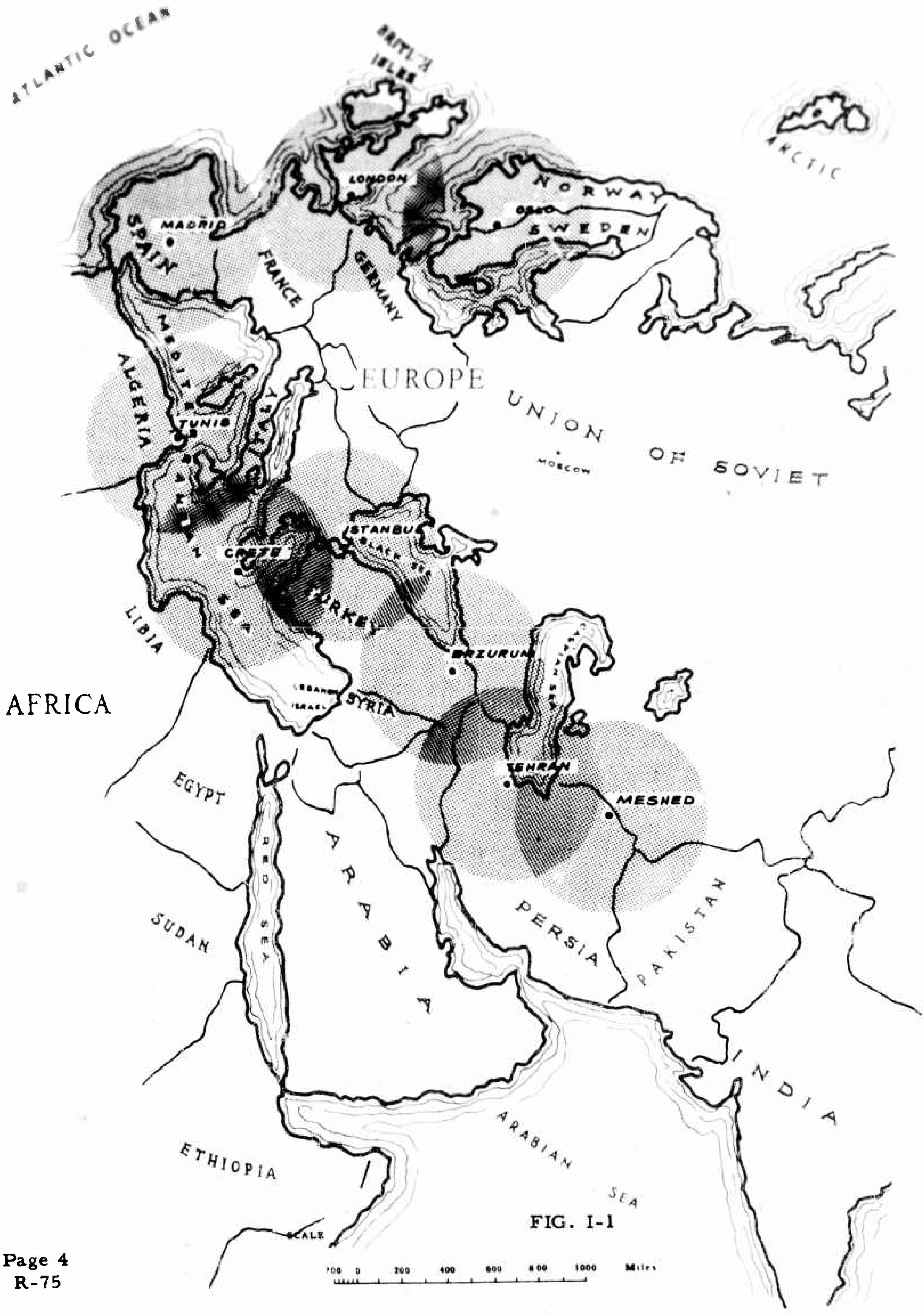
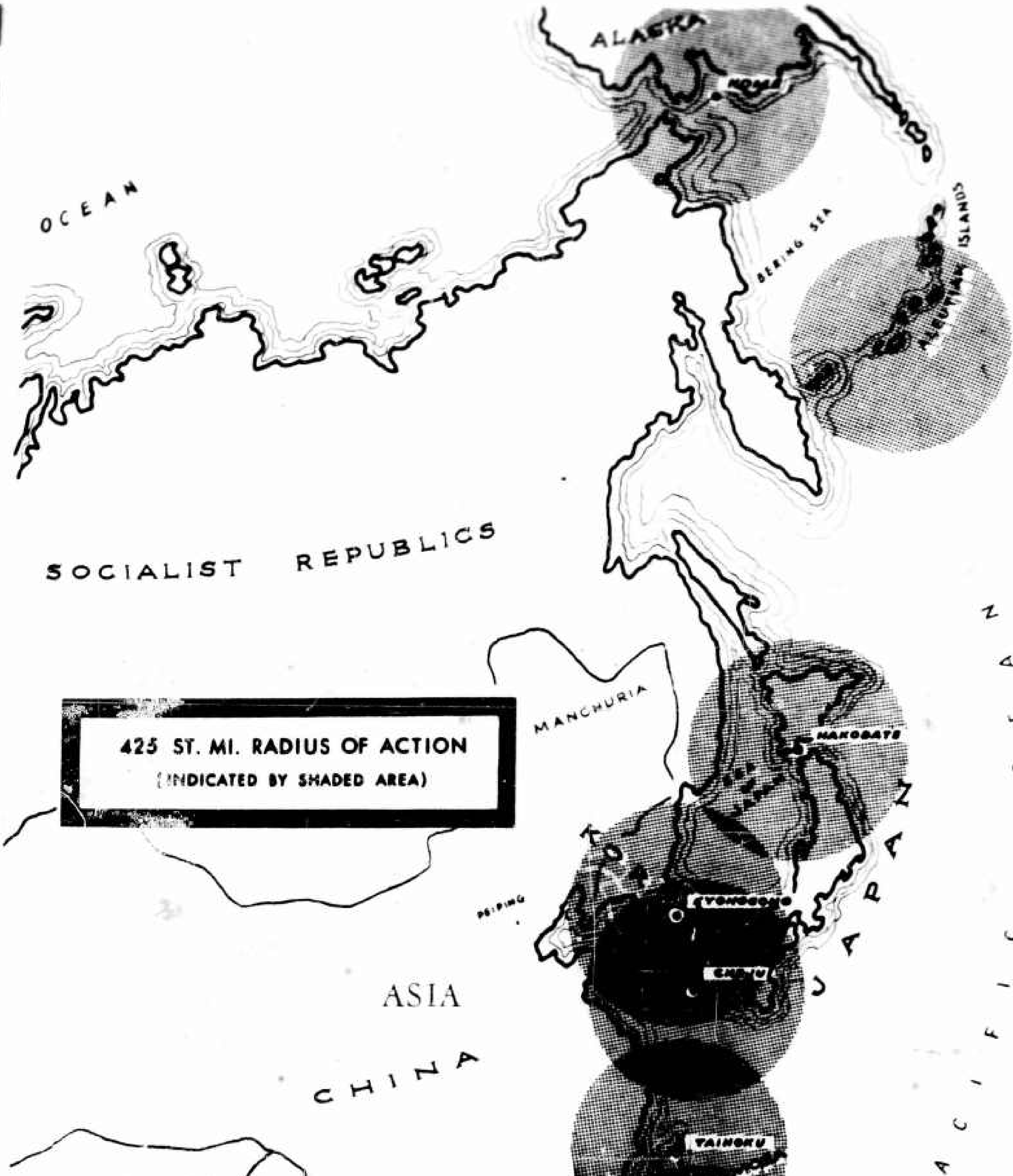

ATLANTIC OCEAN
BRITISH ISLES
ARCTIC
LONDON
NORWAY
OSLO
SWEDEN
MADRID
SPAIN
FRANCE
GERMANY
EUROPE
UNION OF SOVIET
MOSCOW
ALGERIA
TUNIS
ITALY
MEDIT
CRETE
ISTANBUL
BLACK SEA
TURKEY
SEA
LIBIA
ERZURUM
CASPIAN SEA
AFRICA
LEBANON
ISRAEL
SYRIA
TEHRAN
MESHED
EGYPT
RED SEA
ARABIA
PERSIA
PAKISTAN
SUDAN
INDIA
ARABIAN SEA
ETHIOPIA
SCALE
200 0 200 400 600 800 1000 Miles

FIG. I-1


425 ST. MI. RADIUS OF ACTION
(INDICATED BY SHADED AREA)
ALASKA
NOME
BERING SEA
ALEUTIAN ISLANDS
OCEAN
SOCIALIST REPUBLICS
MANCHURIA
HAKODATE
JAPAN
PACIFIC OCEAN
PEIPING
ASIA
CHINA
TAIHOKU
BURMA
INDOCHINA
THAILAND
BINH DINH
PHILIPPINE ISLANDS
South China Sea
BAY OF BENGAL
Origin 40° N

Under item (b), the following concepts were investigated:

1. Tilt-Wing
2. Deflected Thrust
3. Vectored Lift
4. Breguet-Kappus
5. Special Hovering Turbojet
6. Tilting Ducted Propeller

In addition, Dr. Lippisch's "Aerodyne" concept for VTOL was also evaluated under Item (b). In all cases, various types of powerplants were considered where applicable.

D. Results

The results of the Phase I study are presented graphically (Figure 1-2) in terms of take-off gross weight required to meet the mission specifications as a function of cruise speed. It should be noted that the purpose of this phase was to determine the approximate competitive position of the various VTOL design concepts. Therefore, this study was prepared to determine trends of take-off gross weight with speed. This was accomplished through a parametric analysis, taking into consideration both the weight and aerodynamic aspects of the problem. Although the data used in this investigation are believed to indicate correct trends, the absolute values of take-off gross weight are conservative (heavy) due to several basic assumptions. Consequently, the magnitude of gross weight should be used only as a means of a relative comparison between the various design concepts and not construed to be absolute values. In order to obtain exact values of minimum gross weight of aircraft capable to perform the mission, a detailed performance and weight analysis will be made for the more promising design concepts.

Of the many VTOL transport concepts investigated in this Phase I study, the following designs appear to be the most suitable for fulfilling the mission requirements at cruising speeds of 300 mph or greater:

1. Tilt-Wing Propeller
2. Tilting Ducted Propeller
3. Vectored Lift
4. Special Hovering Turbojet
5. Breguet-Kappus
6. Aerodyne

From an evaluation of weights and performance, the tilt wing propeller, tilting ducted propeller and vectored lift designs have approximately equal capability at the specified cruise speed of 300 mph.

For true VTOL operation, the vectored lift concept will always be at somewhat of a performance disadvantage due to the losses in thrust that are accompanied with deflecting the slipstream through quite large angles. At a given gross weight, the loss in thrust requires a greater power which is reflected mainly in increased power plant weight and to a lesser degree in increased propeller weight. Consequently, assuming equal design proficiency, the vectored lift concept will be somewhat heavier than either the tilting ducted propeller or tilt wing designs for VTOL applications.

The results of this study indicate that the required gross weight for the tilt wing propeller and tilting ducted propeller concepts are very nearly the same. From Fig. I-2, the tilt wing propeller has a very slight advantage over the tilting ducted propeller concept. The weight advantage results primarily from decreased required cruising fuel. The shroud drag accounts for the slightly increased fuel requirements for the tilting ducted propeller design.

At higher cruising speeds, the special hovering turbojet, Breguet-Kappus and Aerodyne VTOL concepts become more promising.

Although the results presented in Fig. I-2 indicate the special hovering turbojet aircraft to be the lightest configuration for cruise speeds between 350 to 450 mph, there are several disadvantages of this design for the assault transport mission. Perhaps the greatest detriment is the hot exhaust gases blasting downward in the take-off and landing flight conditions. Another drawback is the limited time available that can be spent in the VTOL regime of flight due to the high fuel consumption.

The Aerodyne overtakes the propeller driven VTOL concepts at approximately 400 mph and the special hovering turbojet aircraft at 450 mph. Beyond a speed of 450 mph, the Aerodyne's competitive position is clearly superior to all the other concepts investigated. However, for the size of aircraft considered the optimum propeller disc loading is high and increases with increasing forward speed, resulting in high effective span loadings. Even in cases of partial power failure (for a reasonable number of engines), safe landings are dubious if not impossible. Obviously, one solution to the above problem is to install emergency power plant units. This study did not consider this requirement and does not reflect the additional weight of such units. Further, to provide adequate stability about all axes of flight duplicate electronic devices would be required. This item has not been included in the trend study. Both of these items would increase the required take-off gross weight, consequently decreasing its indicated competitive advantage. It should also be noted that for an overall evaluation, the Aerodyne is best suited for low altitude operation due th the fact that in forward flight it still depends on lift created by very highly loaded thrust generator. This means that the induced power represents a high percentage of the total power. Since this power depends in turn on air density and is lowest at sea level, hence the Aerodyne is relatively better suited for low altitude operations than other concepts. Other high speed VTOL concepts considered in this study would compare more favorably for cruising at higher altitudes. Nevertheless, the Aerodyne is an interesting concept especially suited for high speed low altitude operation.

Taking into consideration the operational difficulties of the special hovering turbojet concept and the safe landing difficulties with partial power failure of the Aerodyne, the Breguet-Kappus VTOL aircraft is promising for high speed assault transport applications. Since the wing area of this design must of necessity be large to accommodate the submerged ducted propellers, the assumption of cruising at sea level is particularly disadvantageous for this concept. In an overall evaluation, it should be realized that the required take-off gross weight for the Breguet-Kappus VTOL concept would decrease more rapidly for cruising at optimum altitude when compared to either the special hovering turbojet or Aerodyne designs. Thus, the competitive position of this concept would be improved if a more detailed analysis of the cruise condition was made.

In conclusion, this type of broad parametric study to indicate trends is extremely desirable to aid in establishing the competitive position of the many conceived VTOL configurations. However, it should be emphasized that such a study reflects only the weight and performance aspects and does not consider design proficiency or the operational and stability and control problems that may be encountered. Consequently, those concepts that appear to be "optimum" may not prove to be entirely suitable in operation. Some of these problems may be resolved through wind tunnel and component testing. However, the advantages and desirability of having flying test beds to prove and explore the principles of the many competitive VTOL configurations cannot be underestimated.

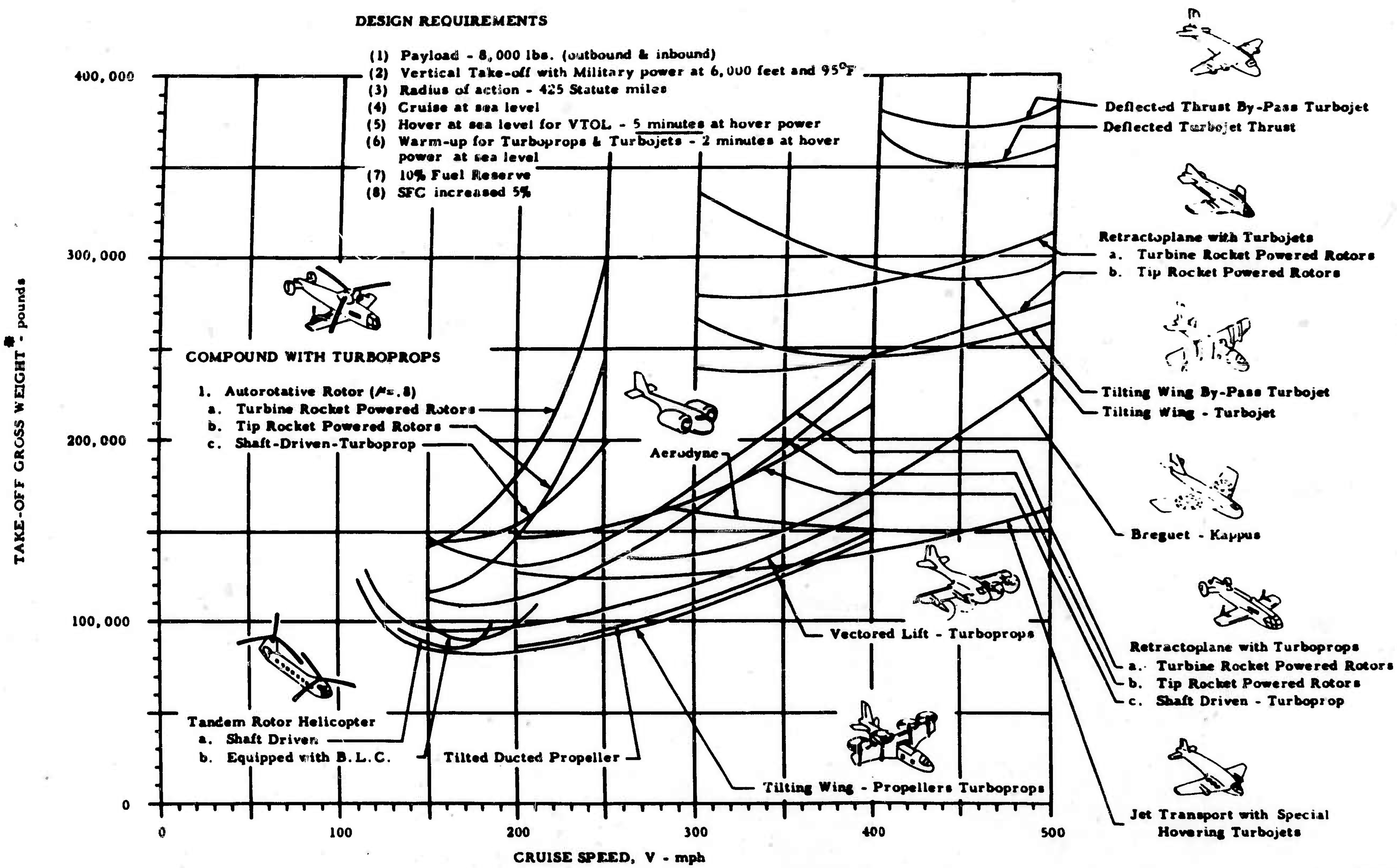
5. Summary Remarks of Take-off Gross Weight Trends (Fig. 1-2)

Configuration	Remarks
a. Rotary Wing Concepts	
1. Conventional Tandem Rotor Helicopter	Cruise speed limited by retreating blade stall. G. W. rise due to rotor weight and increased profile power associated with high solidity (low C_L) requirements for high speed.
2. BLC Tandem Rotor Helicopter	Speed potential improved, G. W. rise delayed.
3. Compound Helicopter	Not suitable for mission as outlined. Probably more competitive if cruising at higher altitudes.
4. Retractoplane	Most competitive of rotary wing concepts at moderate speeds. G. W. high due to heavy rotor and drive systems. For increased time in hovering and high speed potential, turbojet powered retractoplane would probably be more competitive.
b. Fixed Wing Concepts	
1. Tilt-Wing	Tilt-wing propeller optimum aircraft for specified mission. Tilt-wing turbojet heavy due to power plant weight and hover fuel requirements.
2. Deflected Thrust	Not competitive due to losses in thrust deflection resulting in increased power plant weight and hover fuel requirements.
3. Vectored Lift	Competitive with tilt-wing propeller. Heavier due to thrust loss for deflection of slipstream.
4. Breguet-Kappus	Competitive at high speeds. Heavy due to high fuel requirements associated with low wing loadings due to submerged fans. More competitive at higher cruising altitudes.
5. Tilting Ducted Propeller	Competitive at mission speed requirement.
6. Aerodyne	Superior at speeds greater than 450 mph. Disadvantages due to reliance on power for lift and electronic devices for stability. Less competitive at higher cruising altitudes.

VERTOL Aircraft Corporation

COMPARATIVE STUDY OF VARIOUS TYPES OF VTOL AIRCRAFT

TAKE-OFF GROSS WEIGHT * VS CRUISE SPEED



*Values shown represent trends only and are not necessarily the absolute minimum take-off gross weights for the specified mission.

II. INTRODUCTION*

A. MISSION REQUIREMENTS

Development of aircraft capable of vertical take-offs and landings and also capable of much higher flying speeds than contemporary helicopters has received great impetus due to recent developments in the aerodynamics of high lift generation and thermo-propulsion. Such aircraft have been conceived in a number of configurations, the relative advantages of which have not been established on an analytical basis.

In May 1955, Vertol Aircraft Corporation was awarded Contract Nonr 1681(00) from the Office of Naval Research, Department of the Navy, to undertake a broad research comparative study of vertical take-off and landing subsonic transport aircraft in order to analyze and categorize these many design concepts.

At a meeting held at the Office of Naval Research in Washington, D.C. on April 27, 1955, the following mission requirements were set forth:

a)	Payload	8000 lb. out - - - - 4000 lb. back
b)	Take-off	Vertical
c)	Cabin Size	8' x 9' x *
d)	Cargo	35 Infantry troops or equivalent vehicles
e)	T.O. Conditions	Pressure Altitude 6000 ft. at 95°F
f)	Runway Surface	Friction Coefficient μ = .2; UCI = 15**
g)	Cruise Speed	300 mph.
h)	Flight Profile	20% of radius adjacent to target at S.L.
i)	Landing	Vertical
j)	Radius of Action	425 statute miles.

* As Required to accomodate 35 troops.
** Applicable to the case of running T.O. at O.G.W.

Furthermore, it was specified the aircraft must remain controllable with one engine inoperative and be able to make a "controlled crash" landing.

The study is confined to types which offer reasonable technical promise of becoming operationally available within the next 5 to 10 years. Therefore, technical data, such as power plant performance and weights, structural weights, etc., were extrapolated to 1962 state of art.

B. OBJECTIVES

In order to encompass the entire spectrum of VTOL aircraft suitable for a medium payload transport, Vertol Aircraft Corporation suggested that cruise speed be considered as a variable. Consequently, it was agreed that the study be divided into two phases.

* Prepared by: J. Mallen

The phase I study considers all possible design concepts for VTOL transports wherein the take-off gross weight required to perform the specified mission is evaluated and presented as a function of cruise speed. The various configurations studied in the phase I analysis are tabulated in Fig. II-1.

Fig. II - 1 VTOL Concepts

a) Rotary - Wing Concepts

Configurations	Power Plant Hover	Power Plant Cruise
1. Conventional Tandem Rotor Helicopter	Turboprop	Turboprop
2. Tandem Rotor Helicopter equipped with BLC Rotors	Turboprop	Turboprop
3. Compound Helicopter	Turboprop	Turboprop
	Rocket Turbine	Turboprop
	Tip Rocket	Turboprop
4. Retractoplane	Turboprop	Turboprop
	Rocket Turbine	Turboprop
	Tip Rocket	Turboprop
	Rocket Turbine	Turbojet
	Tip Rocket	Turbojet

b) Fixed Wing Concepts

Configurations	Hover	Cruise
1. Tilt Wing	Turbojet	Turbojet
	Turboprop	Turboprop
	By-Pass Turbojet	By-Pass Turbojet
2. Deflected Thrust	Turbojet	Turbojet
	By-Pass Turbojet	By-Pass Turbojet
3. Vectored-Lift	Turboprop	Turboprop
4. Breguet - Kappus	Split-turboprop	Split-turboprop
5. Special Hovering Turbojet	Turbojet (1)	Turbojet
6. Tilting Ducted Propeller	Turboprop	Turboprop
7. Aerodyne	Turboprop (2)	Turboprop (2)

Notes: (1) Special high thrust - light weight hovering engines arranged in clusters.
(2) Shrouded propeller

At the conclusion of this study, the more promising concepts for VTOL transport applications will be subjected to a more detailed design study and evaluated for the specified mission.

C. GENERAL METHOD OF SOLUTION

Since the main purpose of the phase I study is to determine the approximate competitive position of each VTOL concept the method of solution was geared to this purpose. Although the results are not considered sufficiently detailed to establish values of design parameters, they are sufficiently accurate to establish trends.

In any study of this sort there are two aspects to the problem:

a) Determination of component weights
b) Determination of fuel required

In the problem of weight prediction, perhaps the most important consideration is the powerplant since so many of the design concepts depend upon low specific weight power generators. For the purposes of this study, only the gas turbine and turbojet engines were considered as primary sources of power. For hovering flight, however, auxiliary sources of power such as tip rockets and rocket turbines, were considered for those VTOL concepts that derive their vertical thrust from a conventional type of helicopter rotor. After visiting various cognizant government agencies, talking with several representatives of industry and reviewing performance data of present day and future engines, trend curves of specific weight and fuel consumption were determined. These results are discussed in Section V.

Other components of weight empty were based in most part, on actual aircraft data obtained from various government agencies or from our own experience. For each component weight, pertinent design parameters were determined in order to establish the correlating factor. Weight trend data are discussed in Section IV.

In order to expedite the calculations, all weight items were expressed either as a constant number of pounds or in terms of gross weight. However, several of the correlating factors obtained in Section IV & V are functions of parameters other than gross weight. Through manipulation, many of these items could be changed to gross weight. For example, power required may be expressed as gross weight divided by power loading. Power loading in turn can be defined in terms of disc loading. Another example is radius which can be expressed in terms of gross weight and disc loading. These manipulations were made so that only a few of the basic parameters remained to be varied and all the terms were finally represented as a function of gross weight.

Several terms involved gross weight to a fractional power. These terms were linearized simply by solving for a new constant at an assumed gross weight. In this manner, all the component weight terms were made a function of gross weight. Fuel required for cruising a total distance of 850 statute miles was obtained in terms of gross weight using the power plant data and assuming power required remains constant, i.e. no variation in gross weight was considered as fuel is consumed. Summing all these items and solving for required take-off gross weight was an easy task. A more detailed discussion of the procedure and assumptions used in evaluating take-off gross weight can be found in Section VI.

D. BASIC ASSUMPTIONS

1. Mission Deviations

To evaluate and categorize the many VTOL design concepts as quickly as possible, several deviations from the specified mission were made.

a) Payload - 8000 lbs. outbound and inbound
b) Cruise at sea level
c) Power required to cruise at a given speed is assumed to be supplied by the engine (s) at 80% of its (their) rated military power.

These deviations were made in order to facilitate the calculations and do not effect trends but merely result in conservative (heavy) estimates for take-off gross weight. Item (c), cruise at 80% of rated military power, may be criticized from an operational viewpoint since this assumption requires shutting down a sufficient number of engines to assure the cruise condition.

In the next phase of this study, the tilt-wing propeller and vectored lift VTOL concepts will be optimized in order to minimize the take-off gross weight. One item in the optimization will be the determination of the best cruising altitude and speed for the specified mission taking into consideration part-load fuel consumption characteristics of the power plants.

2. Engine-out Safety Provisions

In order to satisfy the requirement that the aircraft remain controllable with one-engine inoperative and be able to make a "controlled crash" landing, inter connecting shafts have been provided for all applicable design concepts. In addition weight provisions and power requirements have been estimated to assure positive control throughout all regimes of flight with particular emphasis on the take-off and landing conditions.

3. Time in Hovering

Time spent in hovering was not specified. However, this is a very important factor and may have considerable influence in the selection of the optimum design parameters. To illustrate the importance of hovering time, the hourly fuel consumption per pound of vertical thrust is shown in Fig. II-2 for various types of vertical thrust generators, ranging from conventional helicopters to turbojets. In order to correlate on a common basis the relative fuel consumption of various thrust generators, the jet exit area loading has been selected as an independent variable.

Jet exit area loading is defined as the thrust divided by the cross section of the fully developed air stream associated with the generation of thrust. It may be recalled at this point that for the airscrew type of thrust generators, the ultimate jet area loading is equal to twice the disc loading. This results from the assumption that the cross section area of the ultimate slip stream is equal to one-half of the airscrew disc area.

3. Time in Hovering (Continued)

For such thrust generators as ducted propellers and fans, as well as turbojet engines, the exit area of the duct or exhaust pipe may be considered as the "jet exit" area.

In order to provide some feeling as to the efficiency with which thermo-chemical energy of fuel is used in the process of vertical thrust generation, a few lines showing the equivalent efficiency (η_{eq}) are plotted in Fig. II-2. This equivalent efficiency is defined as the product of the overall efficiency (η_{ov}) and the square root of the ratio of the air density in the fully developed slip stream of the thrust generator (ρ_j) to the density of the ambient air (ρ):

$$\eta_{eq} = \eta_{ov}\sqrt{\frac{\rho_j}{\rho_o}}$$

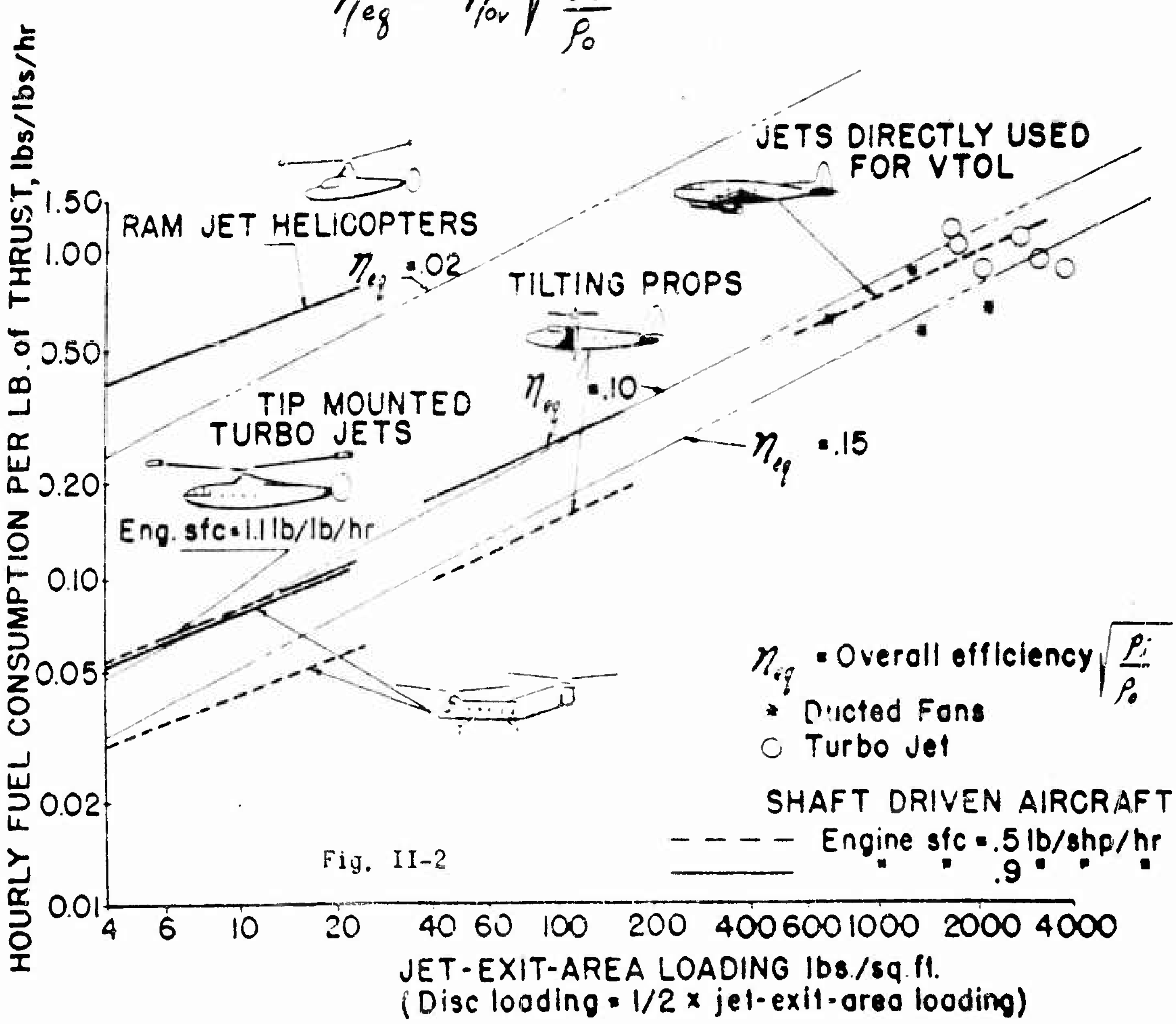


Fig. II-2

APPROXIMATE HOURLY FUEL FLOWS PER LB. OF STATIC THRUST AT S/L

3. Time In Hovering (Continued)

The overall efficiency is defined, in turn, as the ratio of the ideal energy required per unit of time to generate a given thrust to that actually released from the fuel. It can be seen from Fig. II-2, that for such fuels as gasoline and kerosene with an approximate heat value of 19,500 BTU/lb., an equivalent efficiency of 15% may be considered as representative for a very wide range of vertical thrust generators ranging from conventional helicopters to pure turbojets.

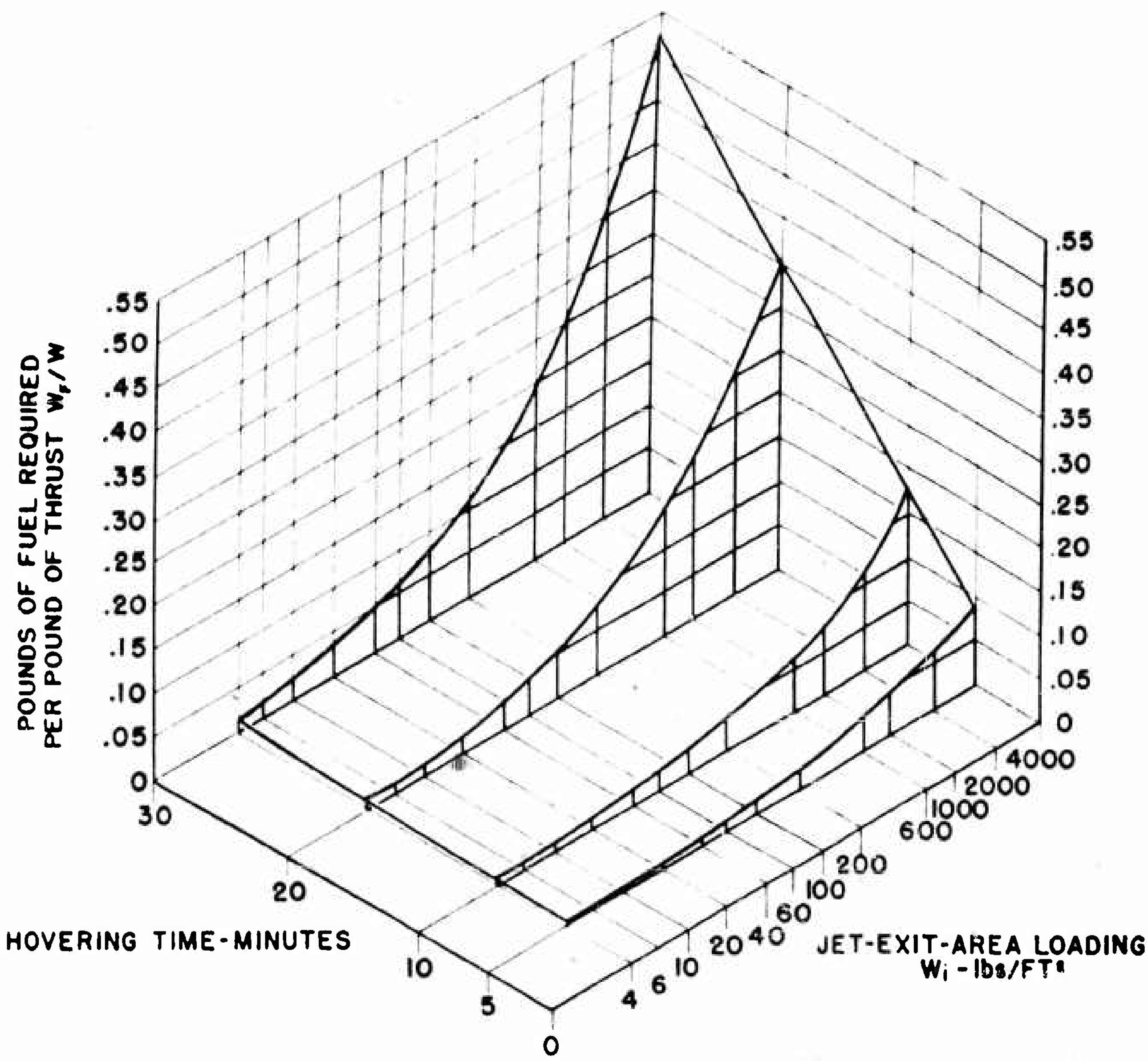


Fig. II-3

APPROXIMATE FUEL REQUIRED PER POUND OF STATIC THRUST AT SEA LEVEL FOR $\eta_{OV}=.15$

3. Time in Hovering (Continued)

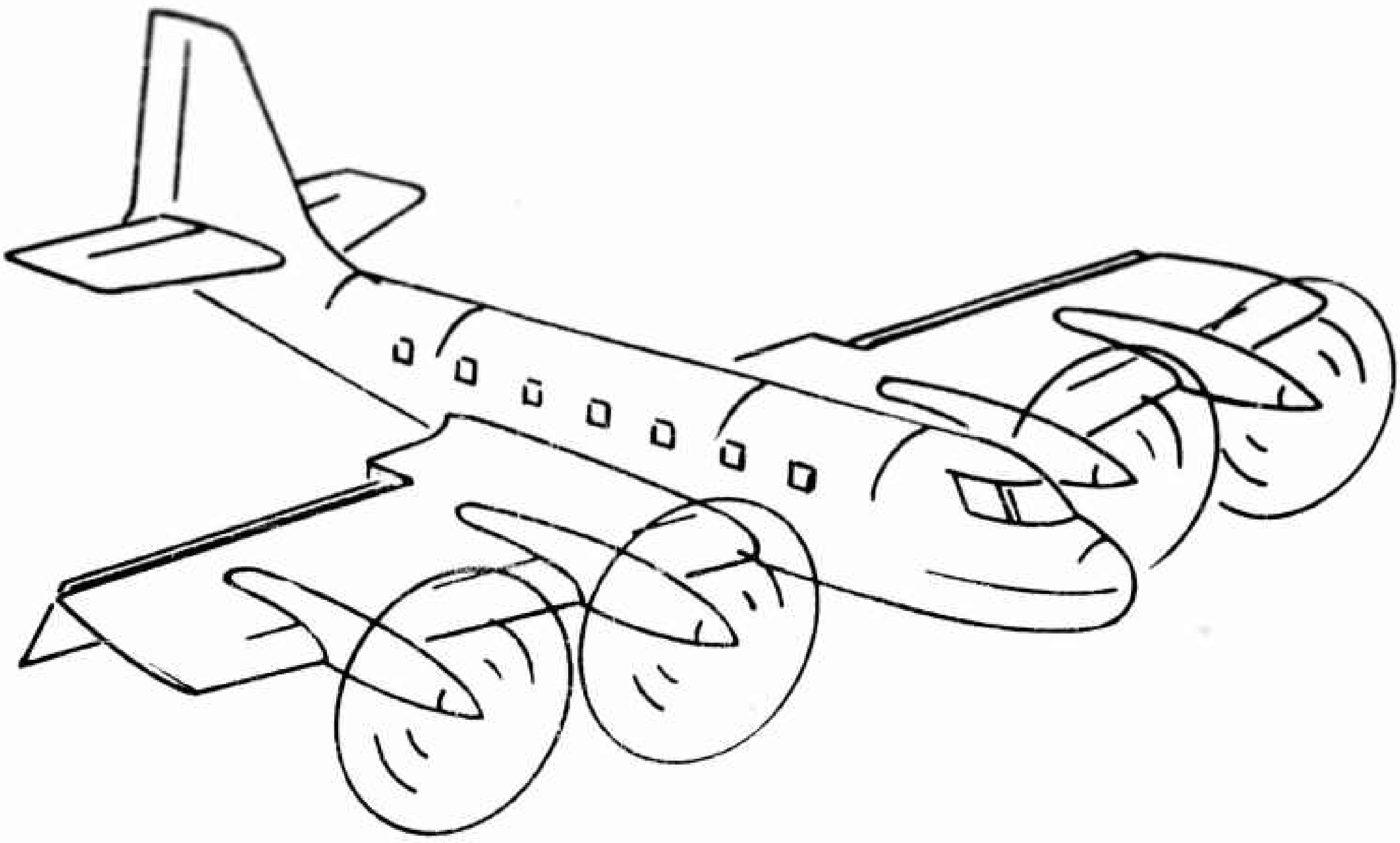
Using this value of equivalent efficiency (η_{o_v} = .15) Fig. II-3 has been prepared showing the approximate fuel required per pound of static thrust as a function of jet exit area loading for several values of hovering time. It can be seen that for extremely short times in hovering (not exceeding 5 minutes), the weight of hovering fuel required, even at very high values of jet exit area loading corresponding to turbojets, is not excessive. However, as the time in hovering is increased, it is apparent that the jet exit area loading must be decreased in order to arrive at reasonable values of hovering fuel required. Consequently, time in hovering is an extremely important factor and will have an appreciably effect in determining not only the optimum design parameters for certain design concepts but will also effect the comparative competitive position of the various VTOL configurations,

To establish a realistic time in hovering to be used in the phase I analysis, a number of operational personnel in the armed forces, government agencies and at VERTOL were approached. The concensus of opinion expressed by these personnel was than an absolute minimum of five (5) minutes would be required to perform the specified mission. In addition to the operational time in hovering, a two (2) minute warm-up was assumed. Therefore, a total time of seven (7) minutes was used in determining fuel requirements in hovering.

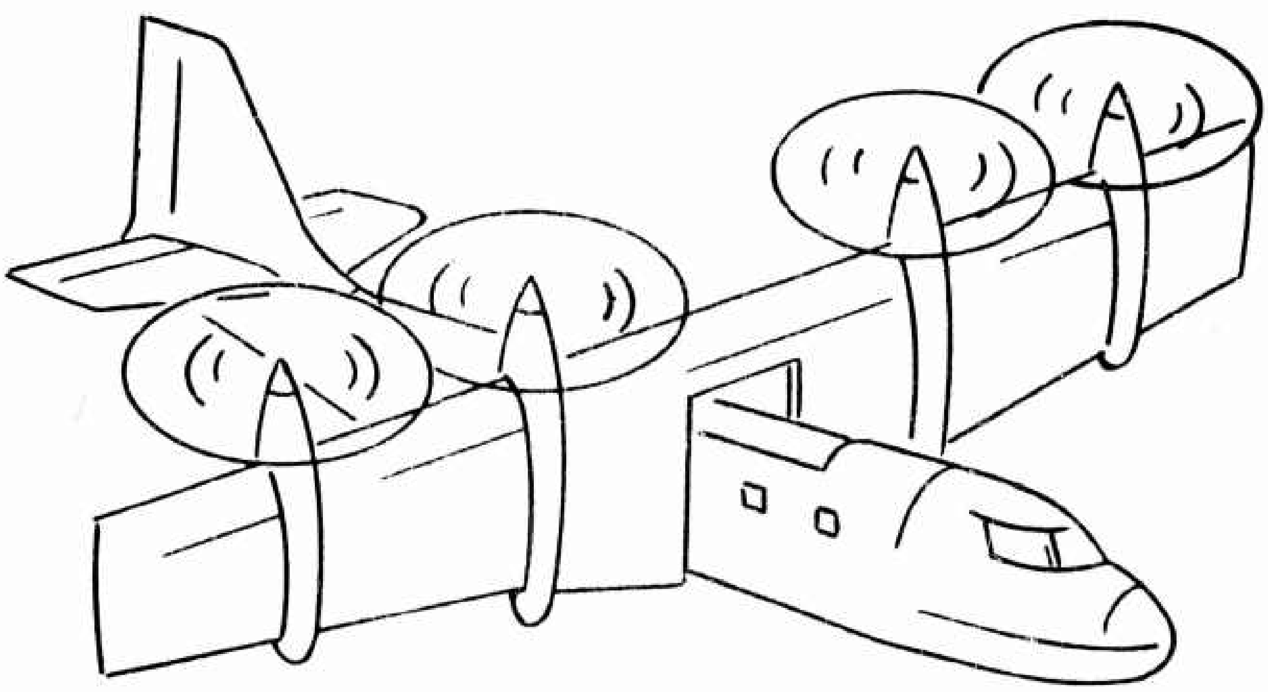
Since hovering time is an important aspect of VTOL operation, the study will be extended in the near future to include the effect of varying time spent in the hovering regime of flight on the competitive position of the VTOL configurations.

4. Design Hovering Altitude

A design hovering altitude of 6000 ft. and 95°F ambient temperature was used throughout this initial investigation. The effect of hovering altitude and ambient temperature on the payload capabilities of the tilt-wing and vectored lift VTOL concepts will be considered in the more detailed study to follow.



VECTORED LIFT - TURBOPROPS CONCEPT



TILTING WING - PROPELLERS TURBOPROPS CONCEPT

III. DESCRIPTION OF VTOL CONCEPTS AND DISCUSSION OF RESULTS*

A. GENERAL

Since some of the design features may not be readily apparent, each VTOL concept investigated in this phase of the work program is briefly described. In addition, for each design concept, charts have been prepared expressing the relative various component weights as a function of cruising speed. To provide a clear presentation of the data, the take-off gross weight was divided into six relative components expressed as percentage of T.O.G.W.:

1. Powerplant Package, W_P. Includes installed engine weight, transmission shafting and drive systems, lubrication and fuel systems.
2. Lift and Propulsive Package, W_L. Includes rotors, propellers and wing.
3. Structural and Fixed Equipment Package, W_Z. Includes body, tail, alighting gear and fixed equipment.
4. Cruising Fuel, W_{CR}.
5. Hovering Fuel, W_{HOV}.
6. Fixed Useful Load, W_{FUL}. Includes crew, payload, trapped liquids, engine oil, etc.

B. ROTARY WING CONCEPTS

1. Conventional Tandem Rotor Helicopter

A conventional shaft driven tandem rotor helicopter, powered with turboprop engines, was analyzed in order to encompass the entire VTOL speed spectrum. This aircraft is competitive with other more promising VTOL schemes at speeds not exceeding approximately 160 mph. Beyond this cruise speed, the gross weight tends to rise rapidly due to increase in rotor weight and higher cruising powers associated with the high solidity - low average rotor lift coefficients required at high speed. At minimum gross weight the disc loading is approximately 4 lbs/sq.ft. and the rotor tip speed is 650 fps. As the cruise speed is increased the optimum disc loading tends to rise and the tip speed decreases.

2. BLC Tandem Rotor Helicopter

To indicate the speed potential of a tandem rotor helicopter, a blowing type BLC system was analyzed. The speed potential of the helicopter was increased by approximately 20 to 40 mph without appreciable increase in take-off gross weight. With more elaborate schemes of BLC, or circulation control potentiality of the helicopter to fulfill the mission can, probably, be extended to still higher speeds.

3. Compound Helicopter

A single rotor helicopter equipped with a wing and propellers for forward flight operation was analyzed. Three separate methods for powering the

* Prepared by: J. Mallen

3. Compound Helicopter (Cont'd)

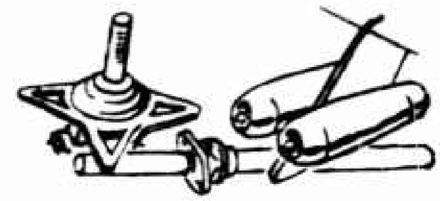
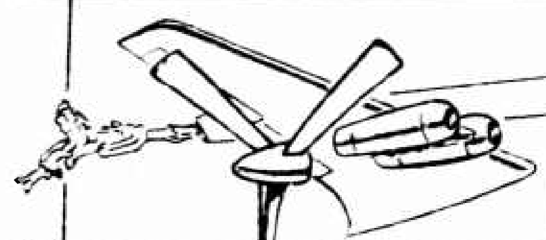
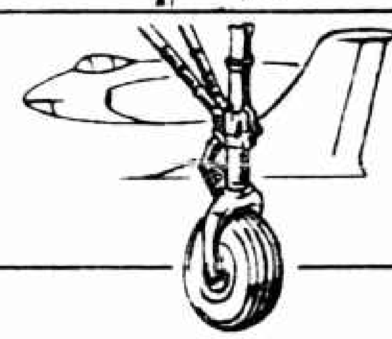
rotor were assumed: shaft drive; tip rocket; and rocket turbine. This was done to make certain that for the specified time in hovering some means other than the shaft drive would prove to be lighter. None of the compound helicopter versions appear to be competitive for this mission. Duplication of the lift and propulsive systems results in high empty weight. Cruise fuel requirements are high due to the hub drag. Optimum disc loading for the shaft driven versions is approximately 3 lbs/sq.ft.; 4 lbs/ft.^{2} for the tip rocket and 5 lbs/sq.ft. for the rocket turbine version. Optimum wing aspect ratio is approximately 6 and operating wing lift coefficient is .5.

4. Retractoplane

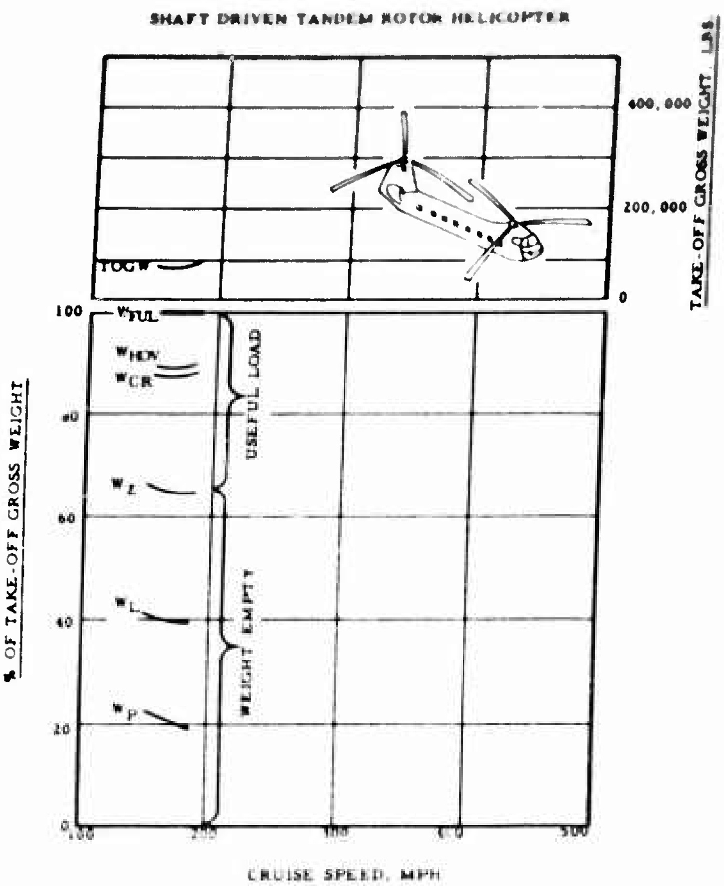
Briefly, the basic concept of the retractoplane is one in which the rotor is telescoped to a small diameter, stopped and retracted into the fuselage after sufficient forward speed has been attained to support the aircraft on wings while forward thrust is provided by propellers or jets. Due to the telescoping and retraction features of the rotor, the lifting system is heavier for the retractoplane than for the conventional or compound helicopter. However, cruising fuel requirements for the retractoplane are considerably less, resulting in lower (than the compound) required gross weight for this concept. The retractoplane is the most competitive of the rotary wing concepts for fulfilling the mission requirements.

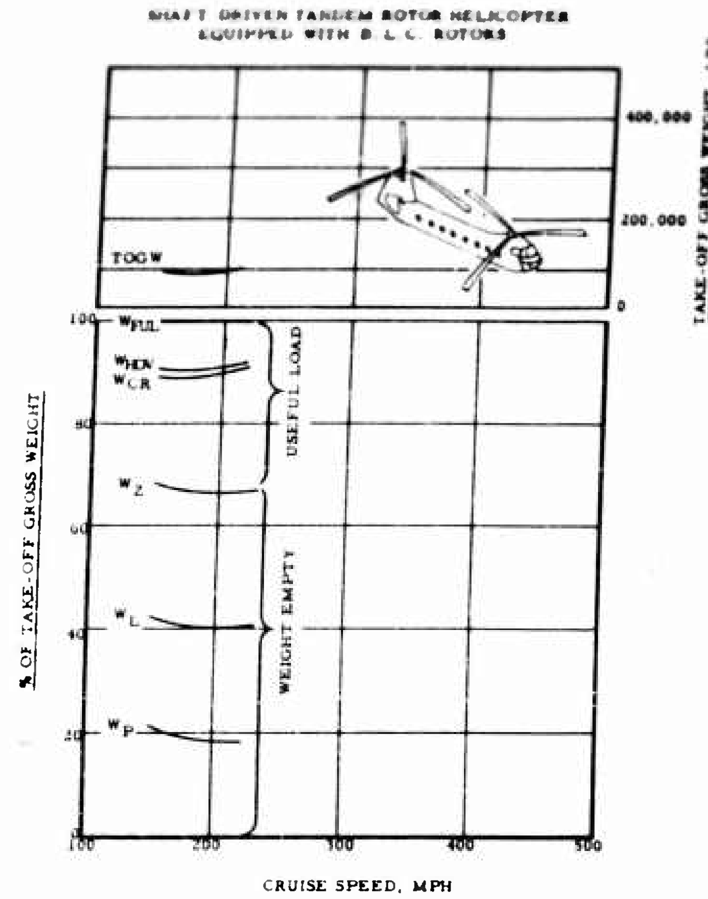
Optimum disc loading for the shaft driven propeller turboprop retractoplane is approximately 6 lbs/sq.ft.; 7 lbs/sq.ft. for the tip rocket version and 8 lbs/sq.ft. for the turbine rocket rotor. A high tip speed low solidity rotor is most suitable. The optimum wing configuration is that of high aspect ratio (AR = 8) and low wing lift coefficient. The requirement for unloading the rotor at moderate forward speeds was primarily responsible for the low operational wing lift coefficients. Consequently, cruising at altitude would be especially beneficial for this design concept.

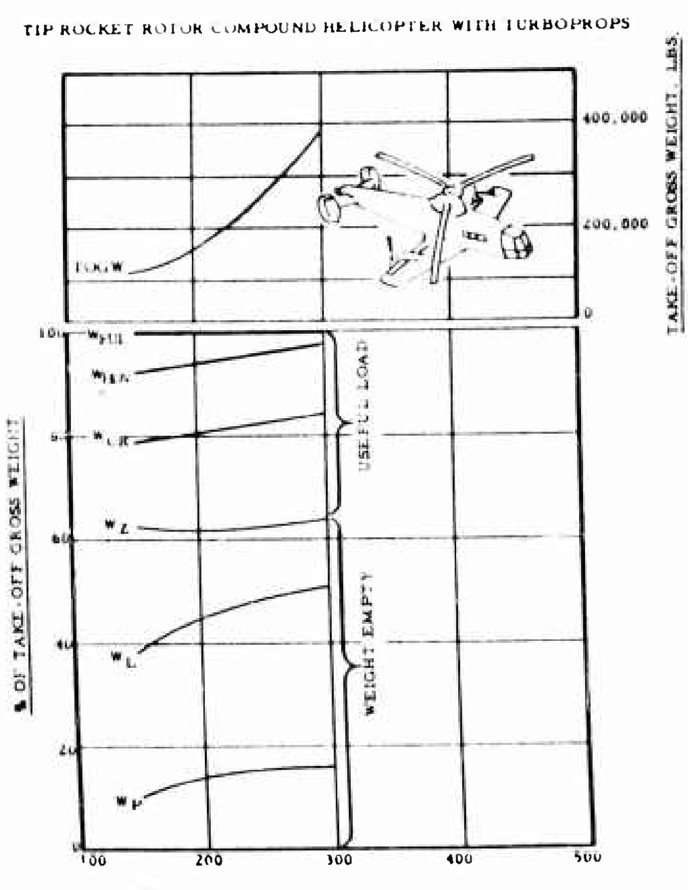
The turbojet versions of the retractoplane are considerably heavier due to the high cruise fuel requirements. Cruising at altitude would reduce the required take-off gross weight. The optimum design parameters are approximately the same as for the turboprop versions.

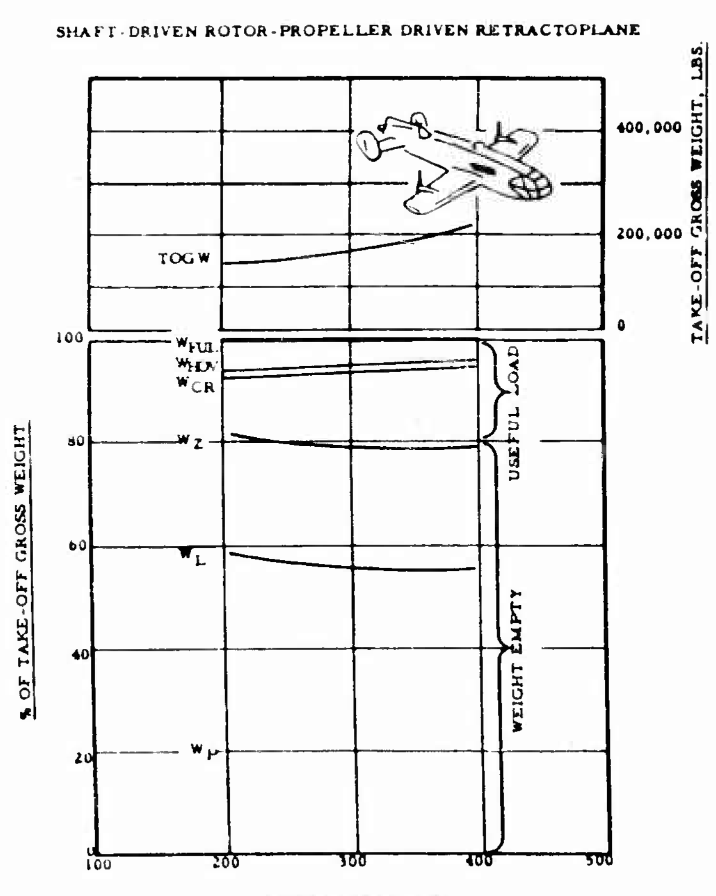
LEGEND		
SYMBOL	ITEM	
W_P	Power Plant Package	
W_L	Lift & Propulsive Package	
W_Z	Structures & Fixed Equipment Package	
W_{CR}	Cruise Fuel	
W_{HOV}	Hovering Fuel	
W_{FUL}	Fixed Useful Load	

CONFIDENTIAL





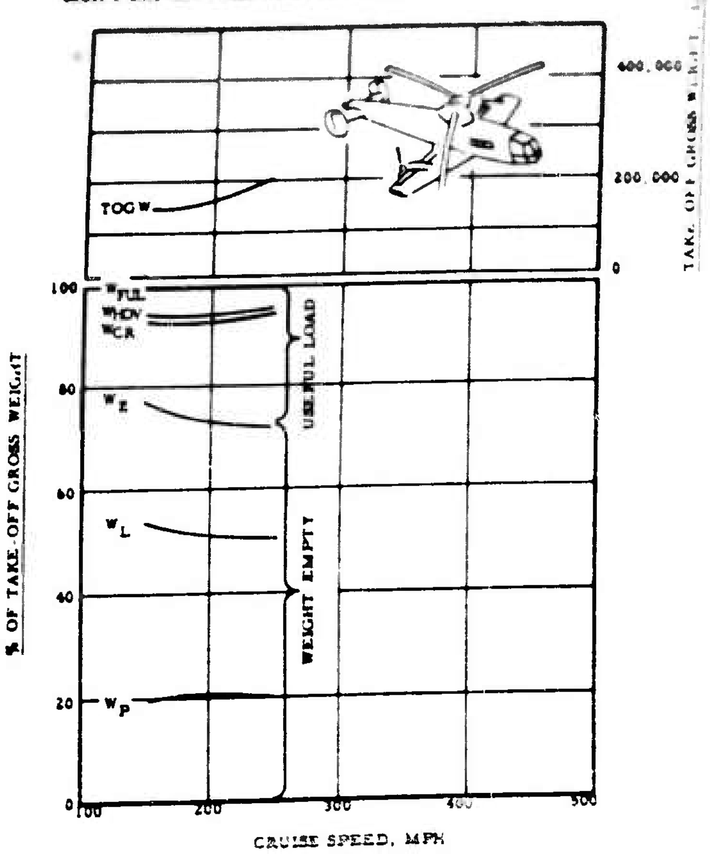




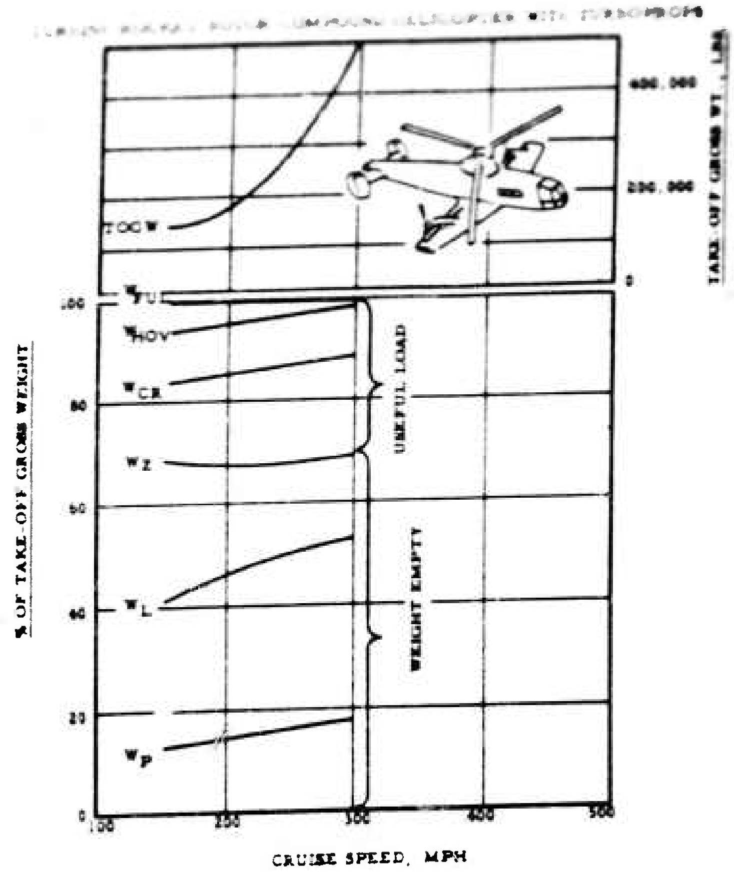
CONFIDENTIAL

ROTARY WING CONCEPTS

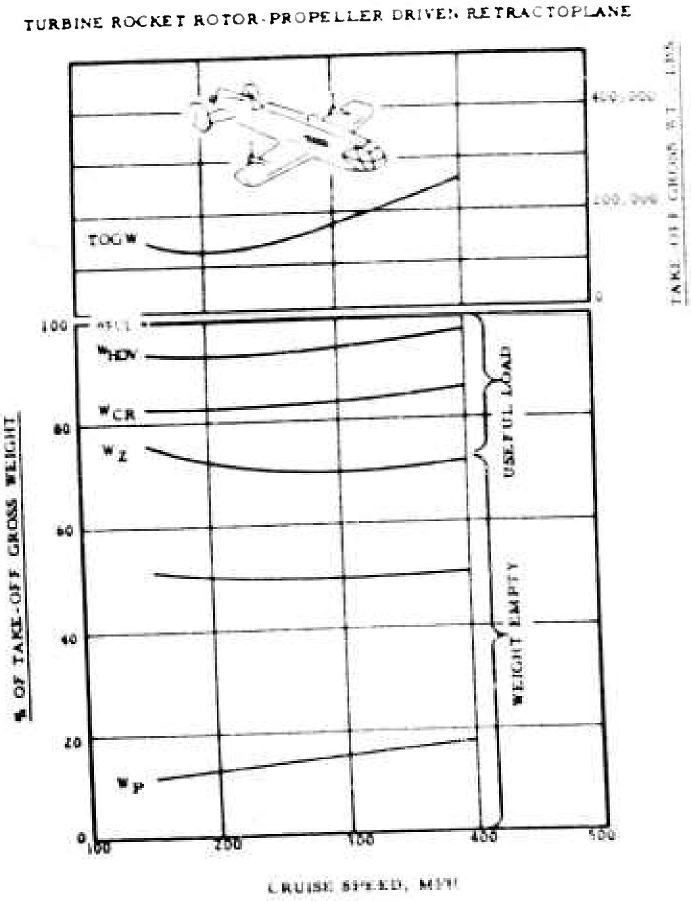
SHAFT DRIVEN COMPOUND HELICOPTER WITH TURBOPROPS



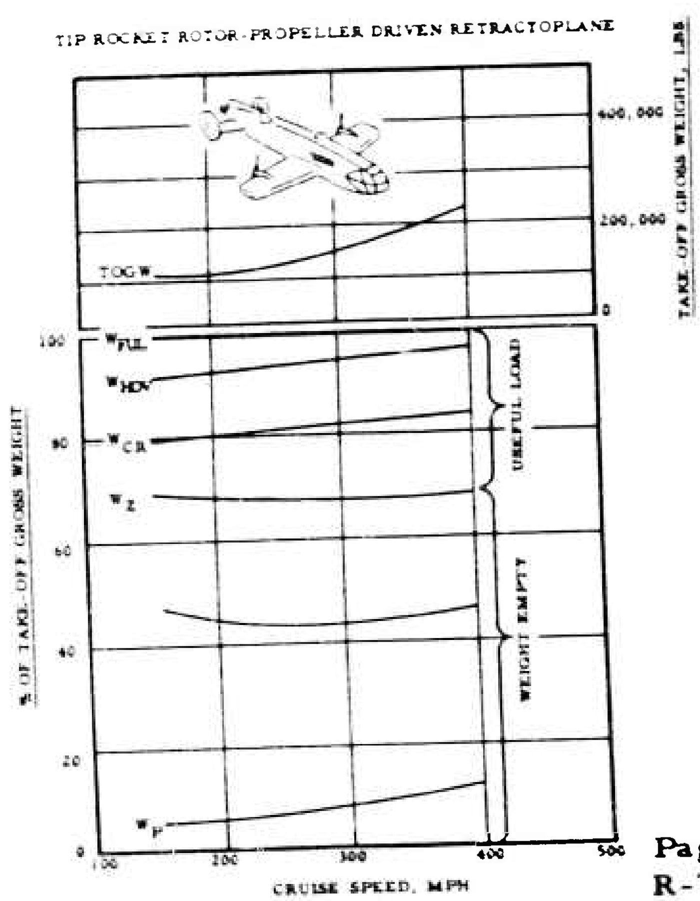
[illegible] COMPOUND HELICOPTER WITH TURBOPROPS



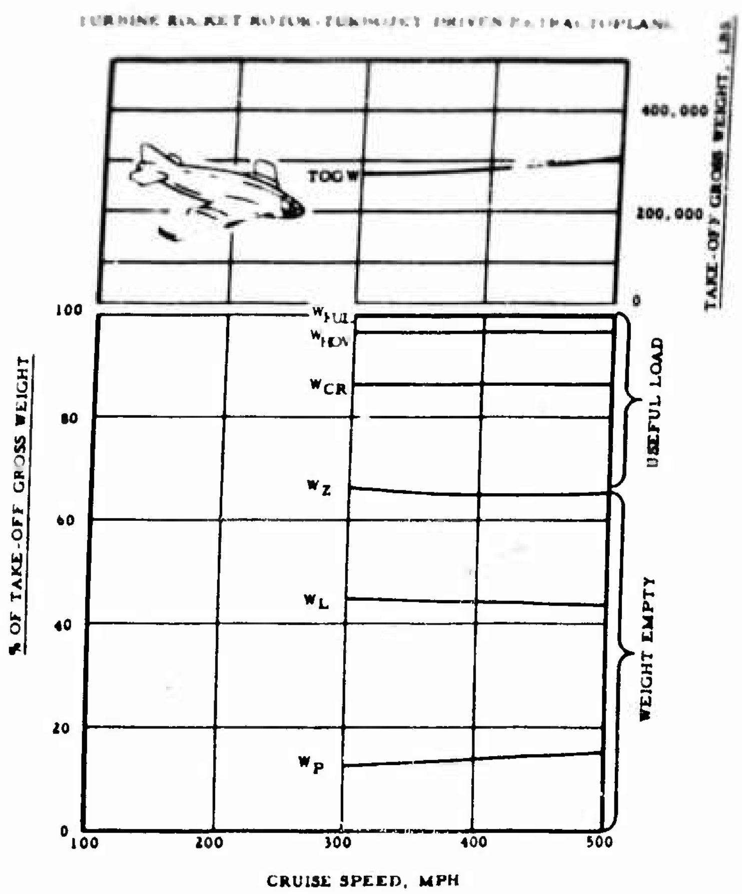
TURBINE ROCKET ROTOR-PROPELLER DRIVEN RETRACTOPLANE

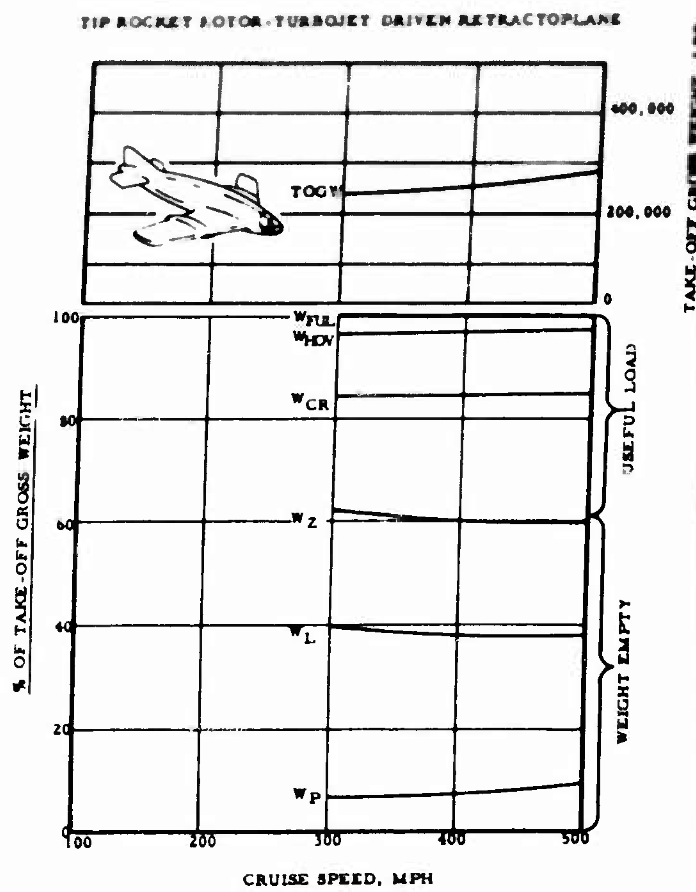


TIP ROCKET ROTOR-PROPELLER DRIVEN RETRACTOPLANE



CONFIDENTIAL





C. FIXED-WING CONCEPTS

1. Tilt-Wing Propeller

The tilting wing propeller concept appears to be the most applicable design in the field of medium speed aircraft wherein the rotor-propellers are used for both lift in hovering and forward flight thrust generation. For this analysis a constant chord wing was assumed, resulting in considerable increase in wing weight. Use of a tapered wing would be reflected in lower values for required take-off gross weight. At the specified mission cruise speed of 300 mph, optimum propeller disc loading is 68 lbs/sq.ft. As forward speed is increased the optimum disc loading increases. A high tip speed high solidity rotor-propeller is required. Wing operating lift coefficients are moderate. The wing span is a function of propeller diameter and number of propellers since the entire wing was assumed to be immersed in the slipstream. Optimum wing aspect ratio is approximately 7.

2. Tilt-Wing Direct Thrust Concepts

The tilting wing turbojet and by-pass turbojet concepts wherein the entire engine-wing package is rotated from near vertical for VTOL to nearly horizontal for forward propulsion are not competitive for this mission. The powerplant size and hovering fuel requirements are primarily responsible for the resulting high values of take-off gross weight. Cruising at altitude would result in a more competitive position for these concepts.

3. Deflected Thrust Concepts

For this design concept, turbojet and/or by-pass turbojet thrust is assumed to be deflected through 90° for VTOL. A 10% loss in thrust was assumed for deflection. Due to this thrust loss, powerplant and hover fuel weight is increased resulting in still higher take-off gross weights. Consequently, these schemes are least attractive for this mission.

4. Vectored Lift

The vectored lift VTOL concept wherein the propeller slipstream is deflected through large angles by means of full-span double flaps for VTOL operation is competitive at the assumed cruise speed. Due to the losses associated with deflection of the slipstream, the vectored lift concept appears somewhat heavier than the tilting-propeller designs for true VTOL applications. Optimum disc loading is approximately 55 lbs/sq.ft. for the 300 mph cruising speed. Wing aspect ratio, assuming Fowler flaps, is about 7.

5. Tilting Ducted Propeller

Vertical take-offs and landings are accomplished by rotating shrouded propellers through approximately 90 degrees. This VTOL concept is competitive at the mission cruising speed. It is slightly heavier than the tilt-wing propeller design due to the increased fuel consumption associated with the shroud drag. Optimum disc loading is approximately 130 lbs/sq.ft., assuming a four-propeller version. Rotation of the inboard ducted propellers through 90° may present some mechanical difficulties.

6. Special Hovering Turbojet

In this design concept, it was assumed that clusters of light-weight turbojets would be mounted vertically for VTOL operations. The turbojet cruise engines would be optimized for the cruising flight. With these assumptions, this VTOL concept is competitive at speeds of 300 mph or greater. However, there are two disadvantages associated with this design for the assault transport mission. First, the hot exhaust gases blasting downward for VTOL would probably create operational problems. Second, due to high fuel consumption associated with these light-weight hovering turbojets, the time spent in the VTOL regime becomes an exceedingly important factor and must be limited to a matter of minutes.

7. Breguet-Kappus

The Breguet-Kappus VTOL concept derives its vertical thrust from ducted propellers submerged in the fixed wings. A hot gas generator of the turbojet type is installed to either drive the ducted propellers through special turbines, or to provide thrust in forward flight, through the discharge of hot gases as in a conventional turbojet. This scheme would obviously necessitate quite large wing areas in order to submerge the propellers. Due to the low wing loadings cruising fuel requirements are quite high for sea level operation. Consequently, for a more realistic mission analysis, the Breguet-Kappus concept for VTOL would be more competitive with the other high speed concepts. Optimum disc loading at 300 mph is approximately 140 lbs/sq.ft. increasing to 200 lbs/sq.ft. at the higher speeds.

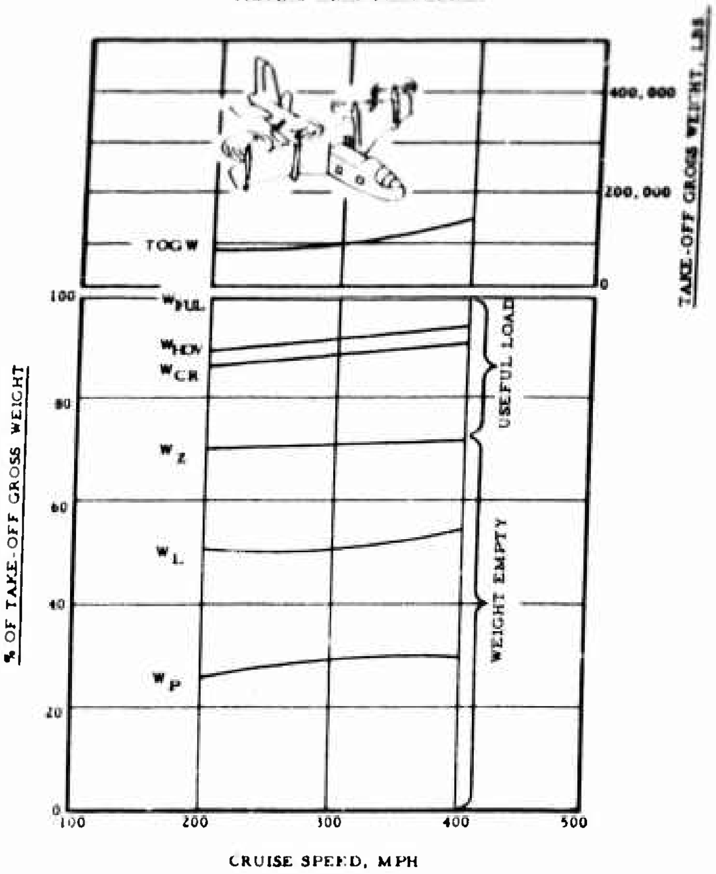
8. Aerodyne

It has been assumed for analysis purposes, that the total thrust (both lift and forward thrust) is derived from the two shrouded propellers. At speeds greater than 450 mph, the Aerodyne's competitive position is clearly superior to all the other concepts investigated. Two disadvantages, however, are associated with this design concept; first, reliance on power for lift and second, dependence upon electronic devices for stability. In an overall evaluation, it should be realized that the Aerodyne is best suited for low altitude operation due to its strong dependence of power required on air density. Consequently, for a more realistic mission analysis, wherein optimum cruising altitude would be selected, the other high speed VTOL concepts would be more competitive with the Aerodyne. At 300 mph the optimum propeller disc loading is approximately 80 lbs/sq.ft. increasing to 275 lbs/sq.ft. at high speed.

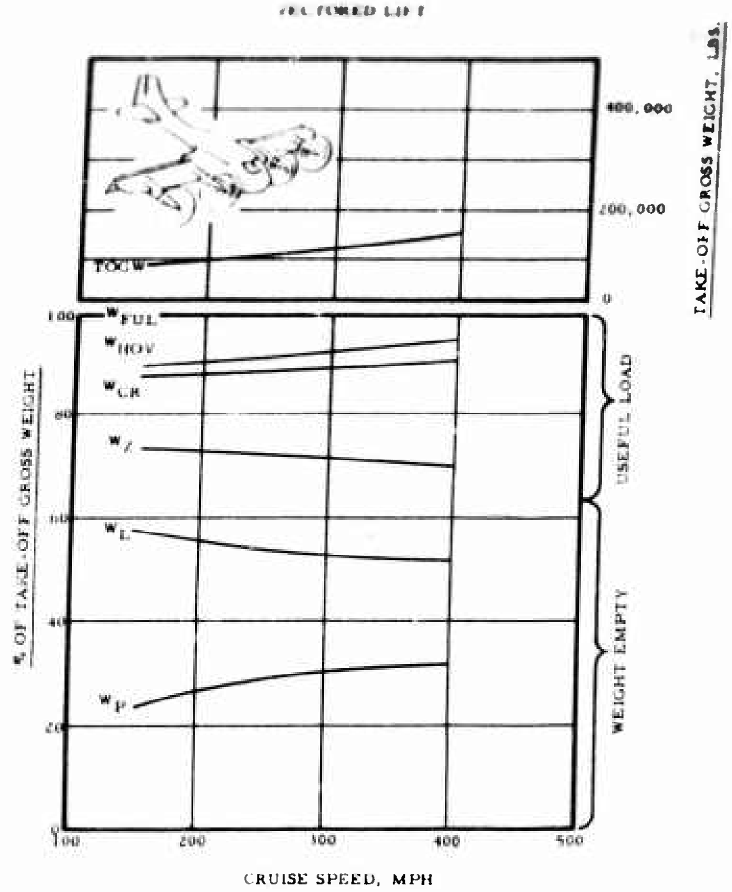
FIXED WING CONCEPTS

CONFIDENTIAL

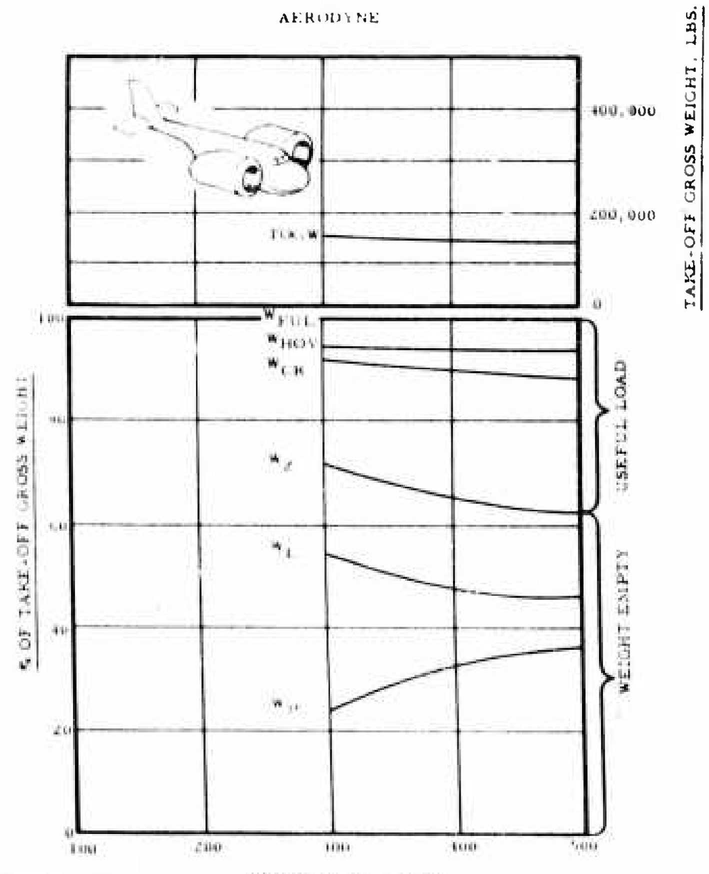
TILTING WING-PROPELLER



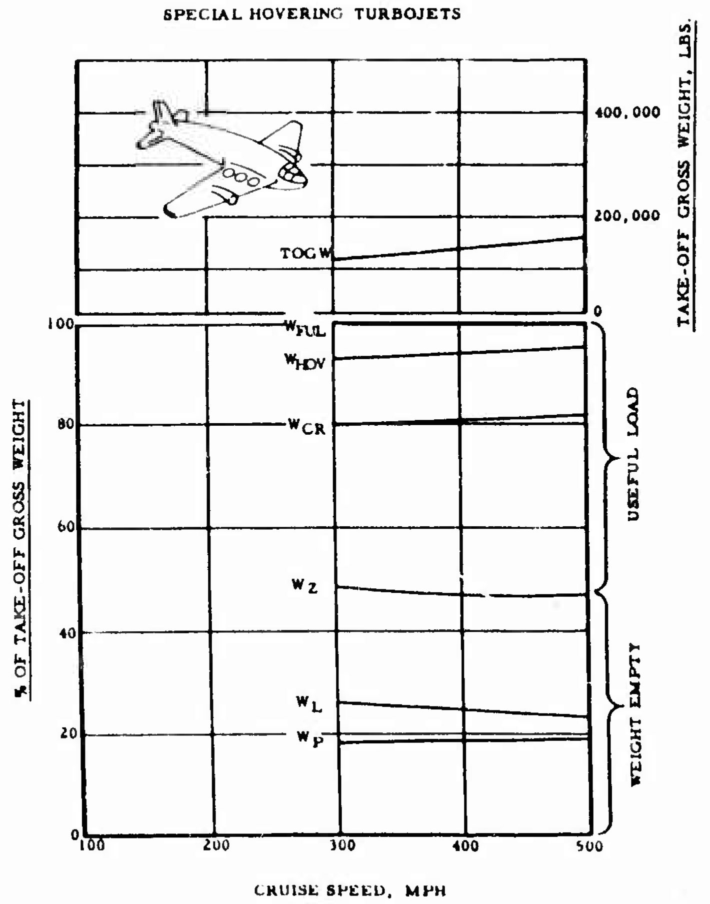
VECTORED LIFT



AERODYNE



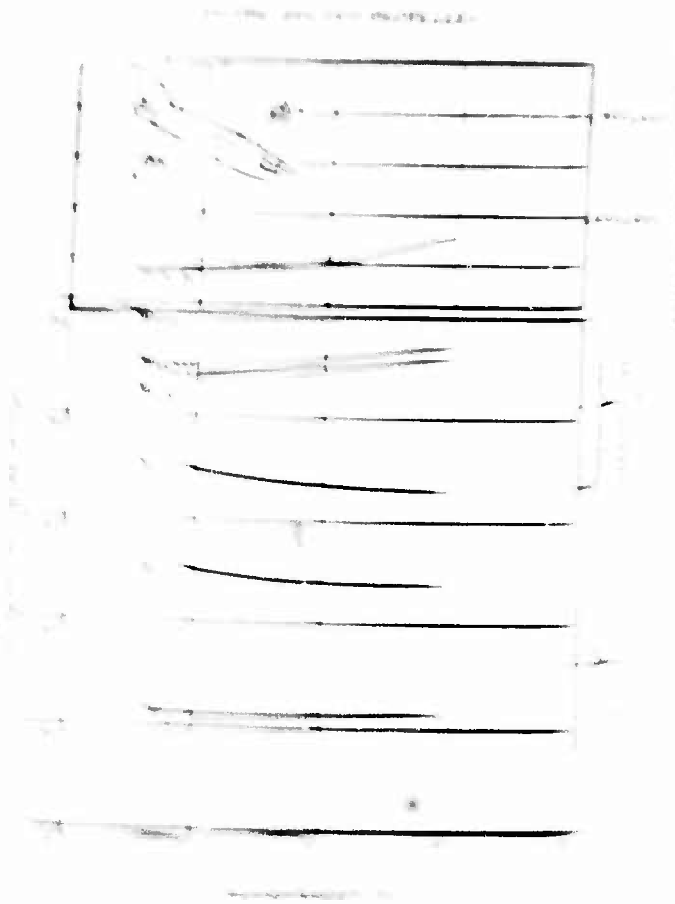
SPECIAL HOVERING TURBOJETS

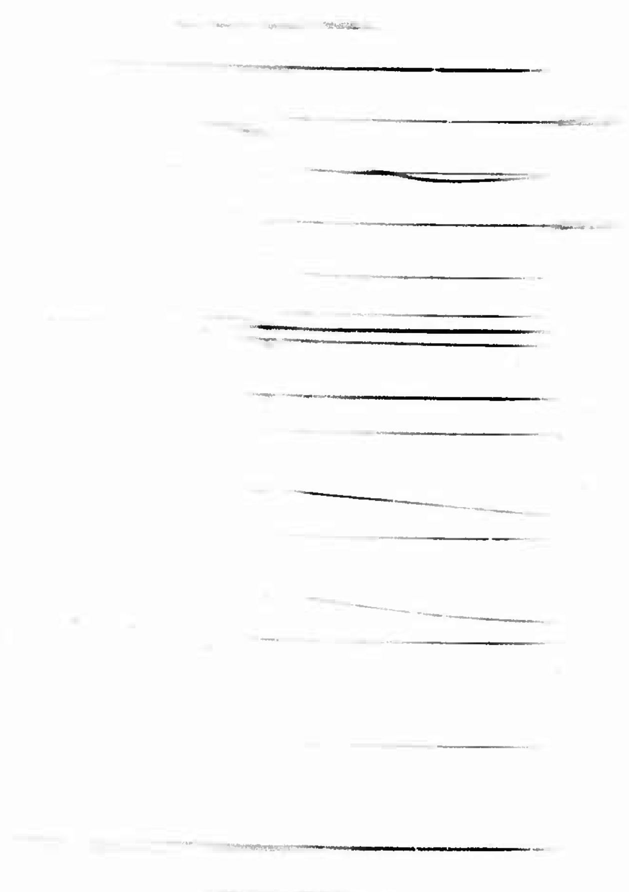


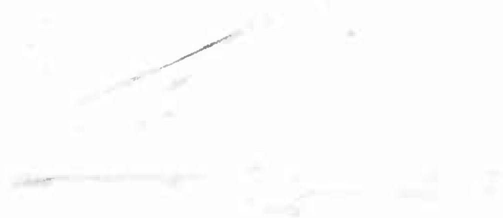
CONFIDENTIAL

FIXED WING CONCEPTS

CONFIDENTIAL

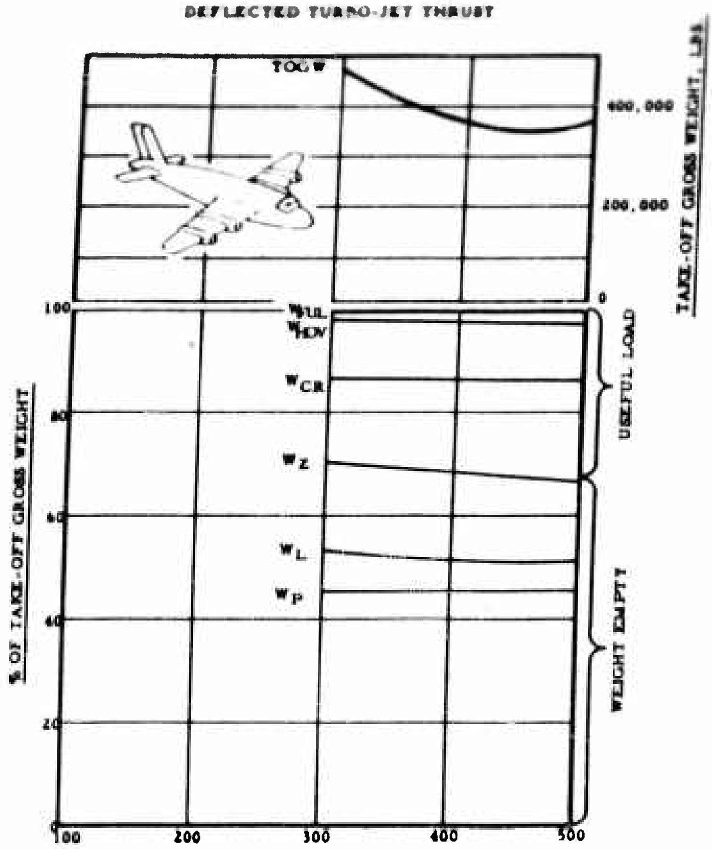




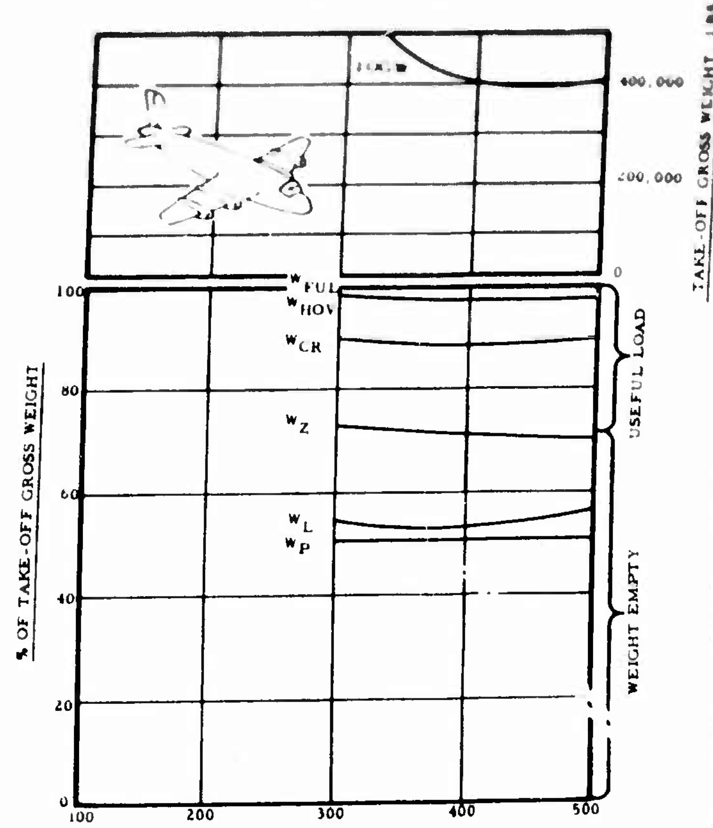


FIXED WING CONCEPTS

DEFLECTED TURBO-JET THRUST



DEFLECTED THRUST BY-PASS TURBOJET



IV. WEIGHT TREND DATA*

A. INTRODUCTION

Since the purpose of this phase of the study is to indicate the competitive position of the various VTOL concepts for a specified mission, the weight analysis has been geared to the prediction of accurate trends rather than detailed absolute values.

Furthermore, in keeping with the purpose of this study, weight trends have been extrapolated to the 1962 "state of art". Consequently, in an attempt to project progress in technology and materials to 1962, the trend weights reported herein are of necessity optimistic as compared to present day data. This approach to the weight analysis is exceedingly important at this stage of investigation, since an inherent problem associated with VTOL aircraft is the high percentage of gross weight represented by weight empty. As a result, in order to perform the specific mission, a concept approaching the borderline of practicality from a weight viewpoint becomes increasingly sensitive to small weight variations in weight empty. Therefore, the prediction of weights should be somewhat optimistic to prevent premature elimination of such a concept from further consideration. The more promis configurations that fulfill the mission requirements will be subjected to a more detailed analysis to confirm and substantiate the accuracy of the initial investi tion.

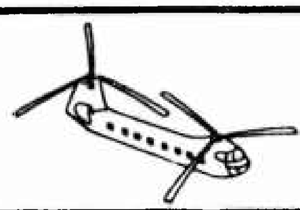
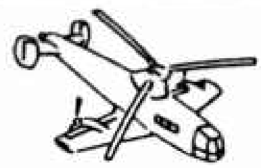
Development of weight expressions for VTOL aircraft has been based on the premise that fixed wing and rotary wing weight trends, with adjustments made to reflect special features and problems, could be combined to predict VTOL weight trends. The design parameters for correlating weight trends have been selected principally for this investigation. The parameters and formulae have been kept as simple as possible, consistent with acceptable accuracy, for ease of calculation and conversion to basic aerodynamic parameters.

The general method of obtaining the weight trends has been to correlate actua weights of various airplanes and helicopters with combinations of basic design pa meters. In most cases, these statistics have been plotted on logarithmic graph p to facilitate obtaining a trend and determining the correlating equation. The tr equations were obtained in most cases by weighting the data considered most repre sentative of efficient design and applying the method of least squares.

In the following sections weight trends are set forth for the various compone groups and applied to the various VTOL concepts. (Summary charts showing the application of these trends follow this page).

*Prepared by: R. Swan

SUMMARY VTOL WEIGHT TRENDS - ROTARY WING CONCEPTS

Item					
	Tandem Helicopter Shaft Driven Rotors		Compound Helicopter Single Rotor with Propellers		
			Shaft Driven Rotor		Tip Rocket Driven Rotor
Item	Conventional Rotor	BLC Rotor	Gas Turbine	Turbine Rocket	
Rotor Group/Rotor Blades	$226K^{0.63}$	$(1.1)(226K^{0.63})$	←	$226K^{0.63}$	→
Hub & Hinge	$92.4K^{0.53}$	$(1.1)(92.4K^{0.53})$	←	$92.4K^{0.53}$	→
Wing Group	—	—	← $1.06\left[41.57C_l\,\omega_W^{0.25}S + \frac{0.6(LF)b^3\omega_W(TF)}{f}\right]$ →		
Tail Group	.01W	.01W	.029W	.029W	.019W
Body Group	← $1.7\left[496K^{0.34}\right]$ →		← $1.26\left[496K^{0.34}\right]$		1.15X →
Alighting Gear*	← .04W →				
Propulsion Group Rotor	.42 #/HP	.42 #/HP	.51 #/HP	.09 #/HP + 360	.145 #/THRUST
Props or Jets	—	—		.51 #/HP	.51 #/HP
Propellers	—	—	← .2 #/HP →		
Drive System Rotor Drive	$610(.6K)^{0.674}$	$610(.6K)^{0.674}$	$305K^{0.674}$	$303K^{0.674}$	ROTOR SHAFT = .01W
Prop. Sync. XMSN	—	—	$130K^{.5}N$	—	—
Prop. Sync. Shafting	—	—	$6.3K^{.5}L$	—	—
Fixed Equipment**	2380 + .03W	2380 + .03W	← 2380 + 0.35W →		
Fixed Useful Load Incl. Eng. Lub. Sys.**	← 9500 →				9050
Fuel & Fuel System Rotor	6.7 #/GAL	6.7 #/GAL	6.65 #/GAL	8.5 #/HP/HOUR	15 #/#THRUST/HR
Props or Jets				6.65 #/GAL	6.65 #/GAL

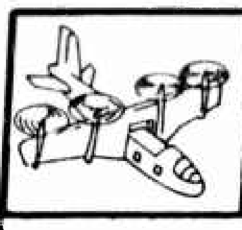
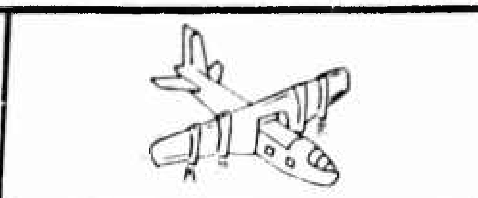
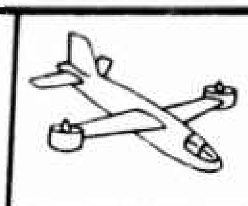
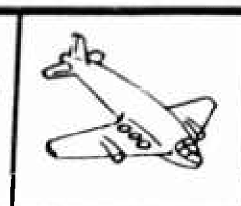
* Retractable - Helicopter Design Criteria

** These values apply only for the gross weight range and mission of this study.

SUMMARY VTOL WEIGHT TRENDS - ROTARY WING CONCEPTS

Retractoplane Single Rotor with Propellers			Retractoplane Single Rotor with Turbojets		Correlation Factor K
Shaft Driven Rotor		Tip Rocket Driven Rotor	Shaft Driven Rotor Turbine Rocket	Tip Rocket Driven Rotor	
Gas Turbine	Turbine Rocket				
←		$(1.2)(226K^{0.63})$		→	$\frac{WR\sigma}{V_T \times 10^2}$
←		$(1.2)(92.4K^{0.63})$		→	$WRHP \times 10^{-7}$
←		$1.06\left[41.57C_1\,\omega_W^{.025}S + \frac{0.6(LF)\,b^3\omega_W(TF)}{f}\right]$		→	
.03W	.03W	.019W	.03W	.019W	
← $1.33\left[496K^{0.34}\right]$ →		$(1.21)(496K^{0.34})$	$(1.33)(496K^{0.34})$	$(1.21)(496K^{0.34})$	$W^2\ S_F \times 10^{-10}$
←		.04W			→
.51 #/HP	.09 #/HP + 360	.145 #/#THRUST	.09 #/HP + 360	.145 #/#THRUST	
	.51 #/HP	.51 #/HP	.277 #/#THRUST	.277 #/#THRUST	
←		.2 #/HP		→	
$270K^{0.674}$	$270K^{0.674}$	ROTOR SHAFT = .01W	$270K^{0.674}$	ROTOR SHAFT = .01W	$\frac{HP}{\Omega}$
$130K^{0.8} \times N$	———	———	———	———	$\frac{HP_x}{\Omega_x}$
$6.5K^{0.8} \times L$	———	———	———	———	$\frac{HP_a}{\Omega_a}$
←		2380 + .043W		→	
9500	9500	9050	9500	9050	
6.65 #/GAL	8.5 #/HP/HOUR	15 #/#THRUST/HR	8.5 #/HP/HOUR	15 #/#THRUST/HR	
	6.65 #/GAL	6.65 #/GAL	6.65 #/GAL	6.65 #/GAL	

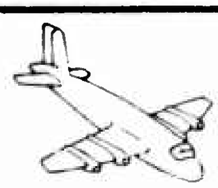
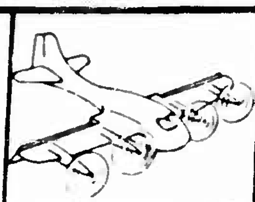
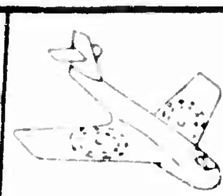
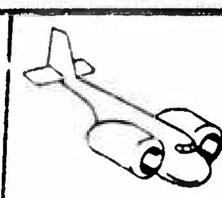
SUMMARY VTOL WEIGHT TRENDS-FIXED WING CONCEPTS

					
	Tilting Wing			Tilting Ducted Propeller	Special Hovering Turbojet
Item	Turboprop	By-Pass Turbojet	Turbojet	Turboprop	
Rotor Group/Rotor	$748K^{0.31}$	INCLUDED IN ENGINE WT.	———	———	———
Wing Group	$X^{1.2}$	$X^{1.2}$	$X^{1.2}$	$X^{1.1}$	$X^{1.0}$
	←		$1.06\left[41.57 C_1 w_w^{.25} S + \frac{0.6(LF)b^3 w_w (TF)}{t}\right]$ →		
Tail Group	.03W ←			→	.03W
Body Group	←		$496 K^{0.34}$		→
Alighting Gear*	.045W	.04W	.04W	.04W	.04W
Propulsion Group Fwd. Flight	.51 #/HP	.306 #/#THRUST	.277 #/#THRUST	.53 #/HP	.277 #/#THRUST
Vert. Flight					.12 #/#THRUST
Propeller/Prop.	———	———	———	$2.0S_S + 1250K^{0.27}$	———
Drive System Prop. Sync. XMSN	$130K^{.5}N$	———	———	$130K^{.5}N$	———
Prop. Sync. Shafting	$6.3K^{.5}L$	———	———	$6.3K^{.5}L$	———
Prop. Extension Shaft	7.2KL	———	———	7.2KL	———
Fixed Equipment**	2380+.026W	2380+.015W	2380+.015W	2380+.025W	2380+.02W
Fixed Useful Load Incl. Eng. Lub. Sys.**	9300	9150	9150	9300	9150
Fuel & Fuel System	←	6.7 #/GAL	→	6.65 #/GAL	6.65 #/GAL
Rotor					

* Retractable - Helicopter Design Criteria

** These values apply only for the gross weight range and mission of this study.

SUMMARY VTOL WEIGHT TRENDS—FIXED WING CONCEPTS

					
Deflected Thrust		Vectored Lift	Breguet-Kappus	Aerodyne	Correlation Factor K
Turbojet	By-Pass Turbojet	Turboprop	Split Turboprop	Turboprop	
———	INCLUDED IN ENGINE WT.	$748K^{0.31}$	———	———	$\left(\frac{HP \times D^3 \times \sigma^{.5}}{\Omega \times 10^4}\right)$
1.0 x ←	1.0 x	1.2 x $1.06\left[41.57 C_1 w_w^{.25} S + \frac{0.6(LF)b^3 w_w(TF)}{t}\right]$	1.30 x	→ $2.5 S_S i$	
.03W ←			→	.03W	
←		$496 K^{0.34}$		→	$W^2 S_F \times 10^{-10}$
.04W	.04W	.05W	.04W	.04W	
.29 #/# THRUST	.345 #/# THRUST	.51 #/HP	.56 #/HP	.53 #/HP	
———	———	———	$1250K^{0.27}$	$1250K^{0.27}$	$\frac{HP \times D^3 \times \sigma^{.5}}{\Omega \times 10^4}$
———	———	←	$130K^{.5}N$	→	$\frac{HP_x}{\Omega_x}$
———	———	←	$6.3K^{.5}L$	→	$\frac{HP_S}{\Omega_S}$
———	———	←	$7.2KL$	→	$\frac{HP_P}{\Omega_P}$
2380 + .015W	2380 + .015W	2380 + .015W	2380 + .025W	2380 + .025W	
9150	9150	9300	9150	9300	
6.65 #/GAL	6.65 #/GAL	←	6.7 #/GAL	→	

B. ROTOR GROUP

1. Rotor Hub and Hinge Weight Trend

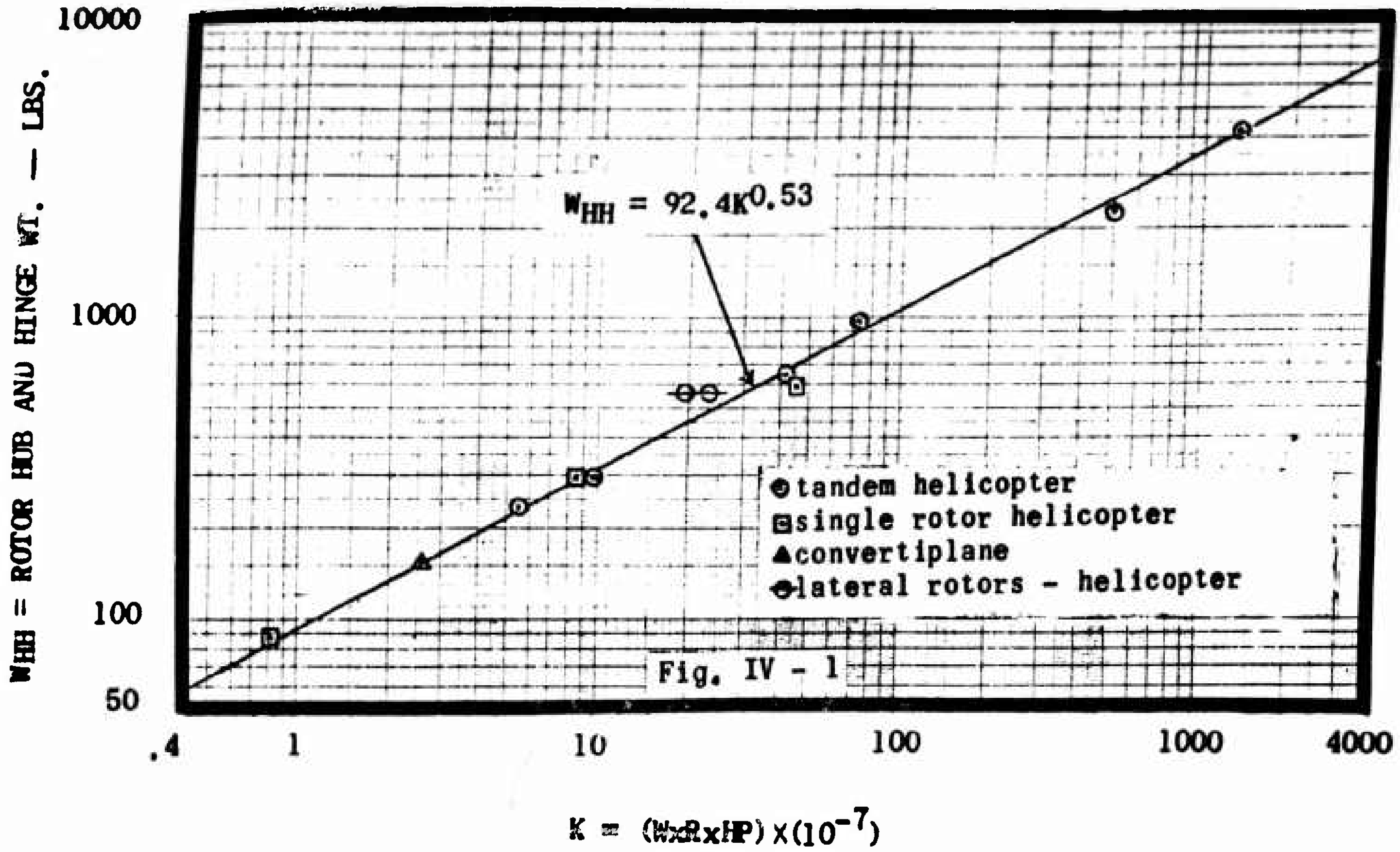


Fig. IV - 1

Three easily determined factors which affect the weight of the rotor hub and hinge are gross weight, torque, and centrifugal force. Assuming that the RPM in torque cancels the effect of RPM in centrifugal force, the factors mentioned above are represented in the correlation factor used in Figure IV- 1 by gross weight, horsepower and rotor radius.

Since the plotted points fall on, or close to, the curve it is logical to assume that the correlation factor contains a reasonable balance of parameters, and that a reliable trend has been established. It is interesting to note that the plotted points include the following variety of rotor systems: teetering two bladed single and tandem rotors; fully articulated four bladed single rotors; and fully articulated three bladed tandem and laterally disposed rotors. Some of the weights were taken from relatively old and obsolete helicopters, particularly the laterally disposed rotors, but the majority are from aircraft presently in use or advanced models.

2. Rotor Blade Weight Trend

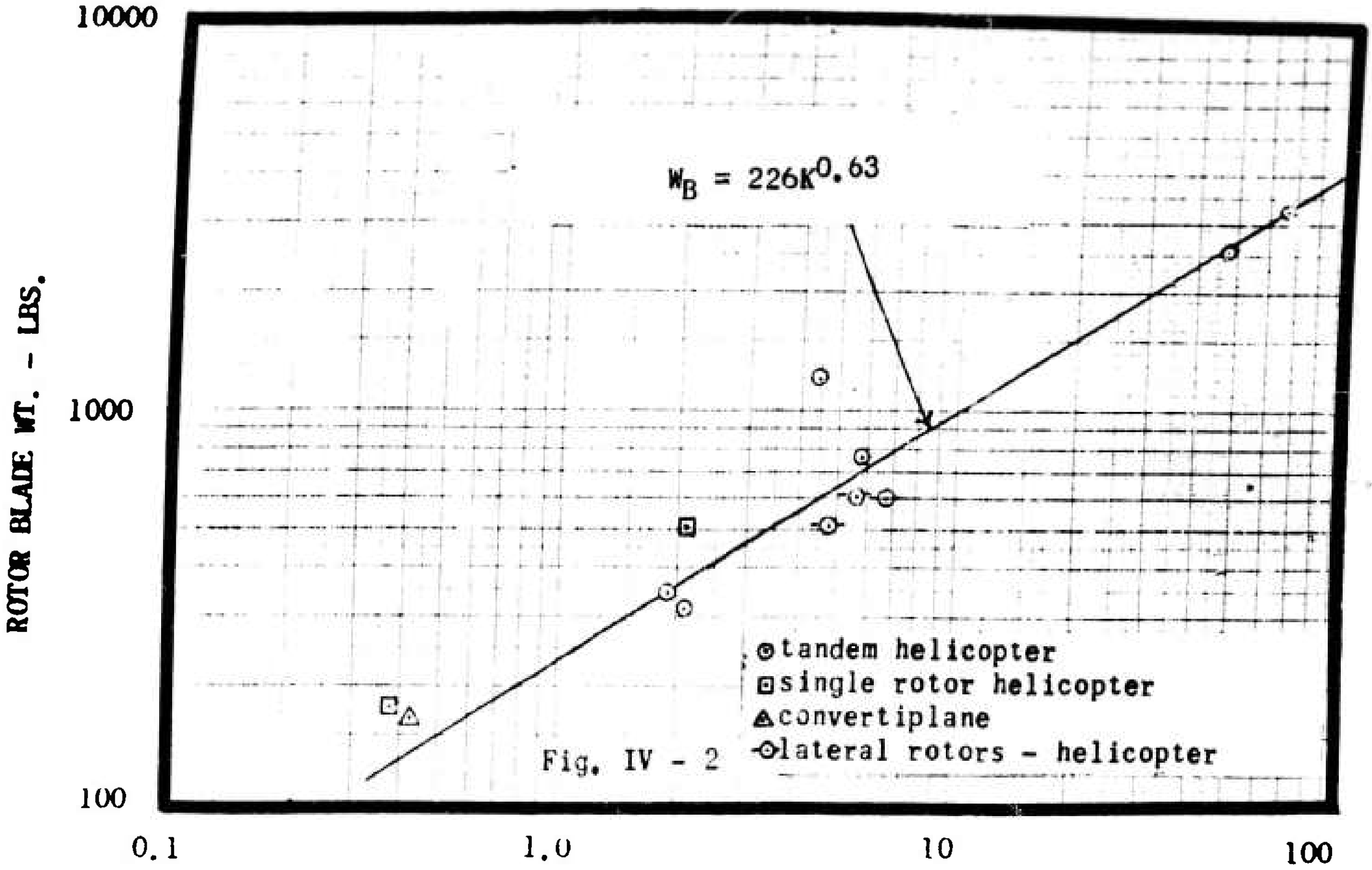


Fig. IV - 2

$$K = (W x R^2 x \sigma) \div (V_T \times 10^2)$$

The rotor blade weight trend indicated by Figure IV-2 represents the best correlation obtained from various combinations of the parameters effecting blade weight. Further study as to the individual effect on blade weight of each parameter within the correlation factor probably would permit a decrease in the scatter of the plotted points; however, the present trend is sufficient for its intended purpose. The wider scatter of blade weight as compared to hub and hinge weight may be attributed to variations in method of blade construction, and more pronounced differences in design philosophy and manufacturing techniques of various manufacturers. For example, two single rotor blades produced by the same manufacturer represent a line approximately parallel to but heavier than the trend curve, while several tandem rotor blades produced by another manufacturer fall on or close to the trend curve.

3. Application of Trends

Conventional Helicopter Rotors

The trend equations were used directly.

Helicopter with Boundary Layer Control

The boundary layer control proposed is of the circulation control or blowing type which would be accomplished by utilizing the centrifugal pumping action of the blades, eliminating the necessity and resulting weight penalty of special blowers. The additional power requirement will be supplied by the helicopters' main engines thus eliminating the need for an auxiliary power unit. Based on a boundary layer control program presently under development by this contractor, it has been estimated that this type can be built into the rotor system for a rotor group weight increase of 10%.

Retractable Rotor

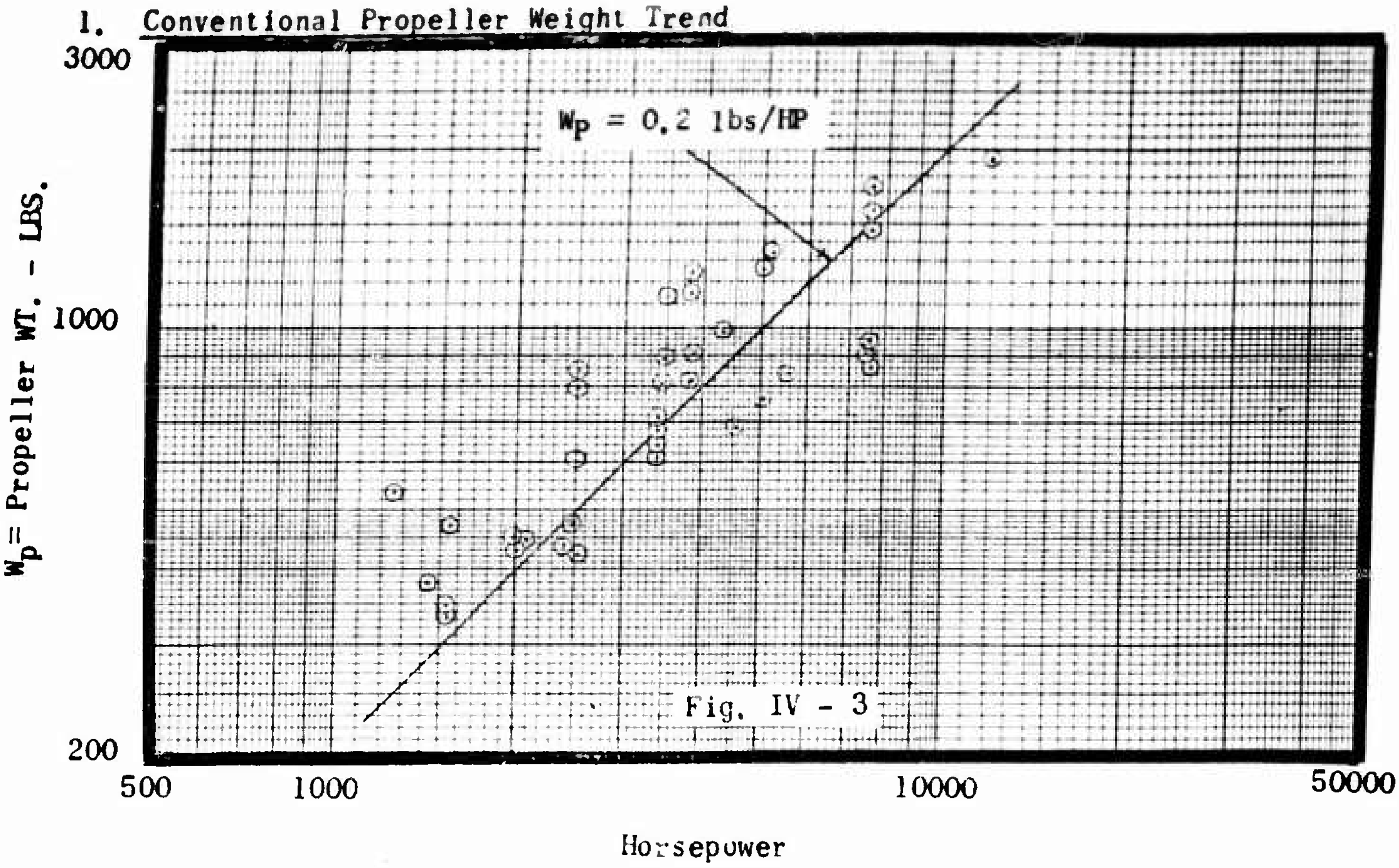
This rotor would consist of a partially telescoping blade which could be retracted inside the fuselage of the helicopter after its rotation has been stopped and sufficient forward speed has been obtained to fully unload the rotor. Preliminary design studies on this type system have shown an estimated weight increase of approximately 38% over the weight predicted by the trend curves. However, in this study the trend curve weight has been increased by only 20% to allow for a more efficient design as the state of the art progresses.

Tip Rocket Driven Rotors

The addition of tip rockets and their fuel lines to a conventional rotor blade would result in a relatively small weight increase. The added complexity of the hub and hinge occasioned by the fuel delivery and control problem would also cause a weight increase. However, the elimination of the rotor driving torque from the hub and hinge would result in a weight decrease. Therefore, it has been assumed that these changes would counterbalance each other and there would be no overall weight change within the accuracy of the study. As a result, the weight from the trend curve has been used without adjustment.

C. PROPELLER

1. Conventional Propeller Weight Trend



The correlation of conventional propeller weight with take-off horsepower was considered sufficiently accurate for this study. Figure IV-3 indicates direct propeller weight variations with horsepower from 0.35 to 0.15 lb. per HP. For use with the compound helicopter and retractoplane versions a value of 0.2 lb. per HP has been used. Although this weight is somewhat below the mathematical average, it is in accordance with the intent of this capability design investigation.

2. Flapping Prop... r Weight Trend

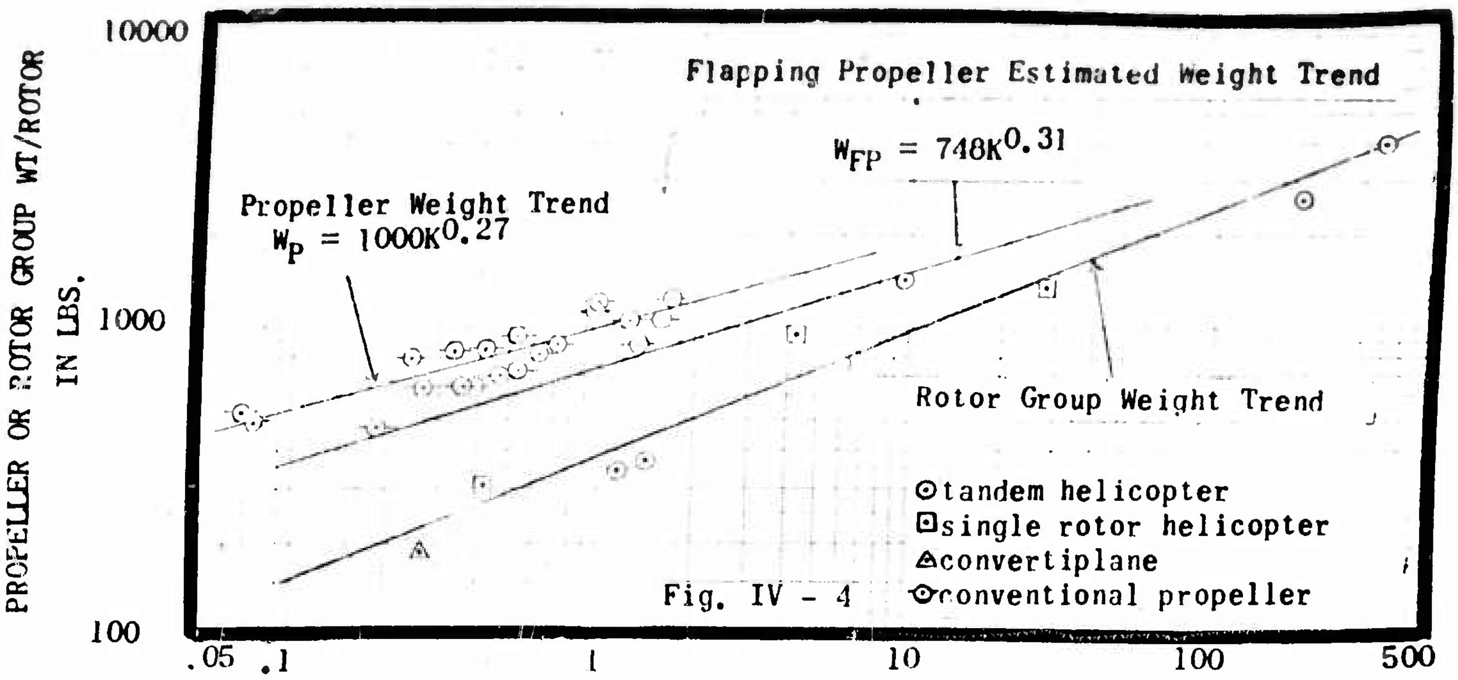


Fig. IV - 4

$$K = (HP \times D^3 \times \sigma^{.5}) \div (\Omega \times 10^4)$$

At the present time there is no weight information available for a flapping propeller of the size and capacity required by the VTOL aircraft under study. In order to estimate a trend for such a propeller both conventional propeller weights including their pitch control mechanism and helicopter rotor group weights per rotor including their upper controls, were plotted against the same correlation factor (Fig.IV-4) In this plot, propellers represent the heavy weight while rotors the light weight extremes. It seems reasonable to assume that the weight of an air screw that is neither a propeller nor a rotor but is a combination of the two, will fall somewhere between the two extremes. The reduction of bending stresses in the propeller blade and hub resulting from utilizing flapping hinges and the elimination of propeller pebble damage criteria, makes possible a lighter system as compared to a conventional propeller. With disc loading much higher than normal for a helicopter rotor the weight probably would tend to increase over that of a conventional helicopter system. Following this reasoning, in the absence of any detailed design data, the flapping propeller trend has been conservatively assumed to fall approximately halfway between the propeller and rotor trends.

This estimated weight trend has been used for VTOL aircraft assumed to have a flapping propeller, such as the tilt-wing and vectored lift concepts.

The tilting ducted propeller, Breguet-Kappus, and the Aerodyne concepts are assumed to utilize non-flapping, high solidity, multi-bladed propellers. The weight trend for a propeller meeting these requirements has been estimated to be equivalent to the conventional propeller trend shown in Fig. IV-4, increased by 25% to reflect a more complicated hub and pitch control mechanism. In addition, the ducted propeller shroud has been estimated to weigh 2.0 lbs. per square foot of shroud area.

D. WING GROUP

1. Wing Weight Trend

Several wing weight prediction methods were investigated in searching for an expression that was simple, reasonably accurate, and applicable to the calculation methods utilized in this trend study. The expression selected which meets these requirements is derived by C.R. Englebry in Ref. 8 and is shown below with some symbols changed to agree with the usage in this report.

$$W_{BW} = 41.57\ C_1 S (\omega_W)^{1/4} + 0.6 (LF) b^3 (\omega_W)\ (TF) + f$$

Where $W_{BW} = W_W$ less ailerons and flaps

An average value for conventional airplane aileron and flap weight is 9% W_{BW}. This percentage has been added to Mr. Englebry's equations for comparison with weights calculated from other expressions and the actual weights for various aircraft as shown in Figure IV-5.

FIGURE IV-5

WING WEIGHT COMPARISION

ITEM	WING WEIGHT				
GROSS WEIGHT SPAN FT ASPECT RATIO	150,000 117 8.5	120,000 141 11.4	64,000 109 8.3	48,700 104 7.6	45,000 93 10.0
	Lbs. %Error*	Lbs. %Error	Lbs. %Error	Lbs. %Error	Lbs. %Error
ENGLEBRY'S (W_{BW} x 1.09)	11,310 +5.3	13,300 -2.5	6,710 -2.5	5,750 -3.3	4,590 -6.1
KELLY'S (From K.D. Wood)	12,800 +19.0	13,100 -4.0	8,066 +17.2	6,381 +7.3	4,949 +1.3
RYAN'S (Based on Kelly's	11,473 +6.8	11,787 -13.6	7,083 +2.9	5,785 -2.7	4,452 -8.9
DRIGGS (Based on K.D. Wood)	10,106 -5.9	10,618 -22.2	6,619 -3.8	5,611 -5.6	3,960 -19.0
MCWHORTER'S	10,290 -4.3	11,939 -12.5	7,100 +3.2	5,918 -0.5	4,735 -3.1
ACTUAL	10,745 0	13,640 0	6,880 0	5,946 0	4,887 0

* Based on Actual Weights.

2. Application of Trends

Assuming that in general a VTOL aircraft will not need the large flap area that a conventional aircraft requires for take-off and landing, only 6% W_{BW} has been added to allow for the VTOL's basic aileron and flap requirements. Therefore, the wing weight is equal to $1.06W_{BW}$. Application of this general rule to the particular cases and some possible exceptions are discussed below.

Compound Helicopter, Retractoplane, and Deflected Jet Thrust

These aircraft will use a conventional wing with only moderate flap requirements, which can be represented by the basic wing weight trend without further adjustment.

Tilting Wing

The basic wing weight has been increased by 20% to allow for the structural requirements, hinges, actuators, and tilting mechanism, necessary to rotate the wing through 90 degrees in converting from vertical to horizontal flight and vice versa.

Tilting Ducted Propeller

An increase of 10% W_{BW} has been estimated to reflect the necessary structural beef-up and complication to permit tilting the propellers.

Vectored Lift

To realize vertical flight by means of vectored slipstream requires large and complex double flaps and possibly a supplementary cascade arrangement. It has been assumed that this can be done for an increase of 20% W_W.

Breguet-Kappus

This system requires large cutouts in the wings which must result in a weight penalty. This weight increase caused by structural beef-up and complication is represented by an allowance of 30%W_W.

Aerodyne

The weight of the shroud and vanes, which replace conventional wings in this aircraft, has been assumed to be 2.5 pounds per square foot of shroud surface area.

E. TAIL GROUP

1. Helicopter Tail Weight Trend

A review of present day tandem rotor helicopters has revealed that an adequate horizontal and vertical tail may be built for 1% of gross weight. A single rotor helicopter tail weight, including the anti-torque rotor and drive system as well as the stabilizing surfaces averages out to about 2% of gross weight.

2. Airplanes Tail Weight Trend

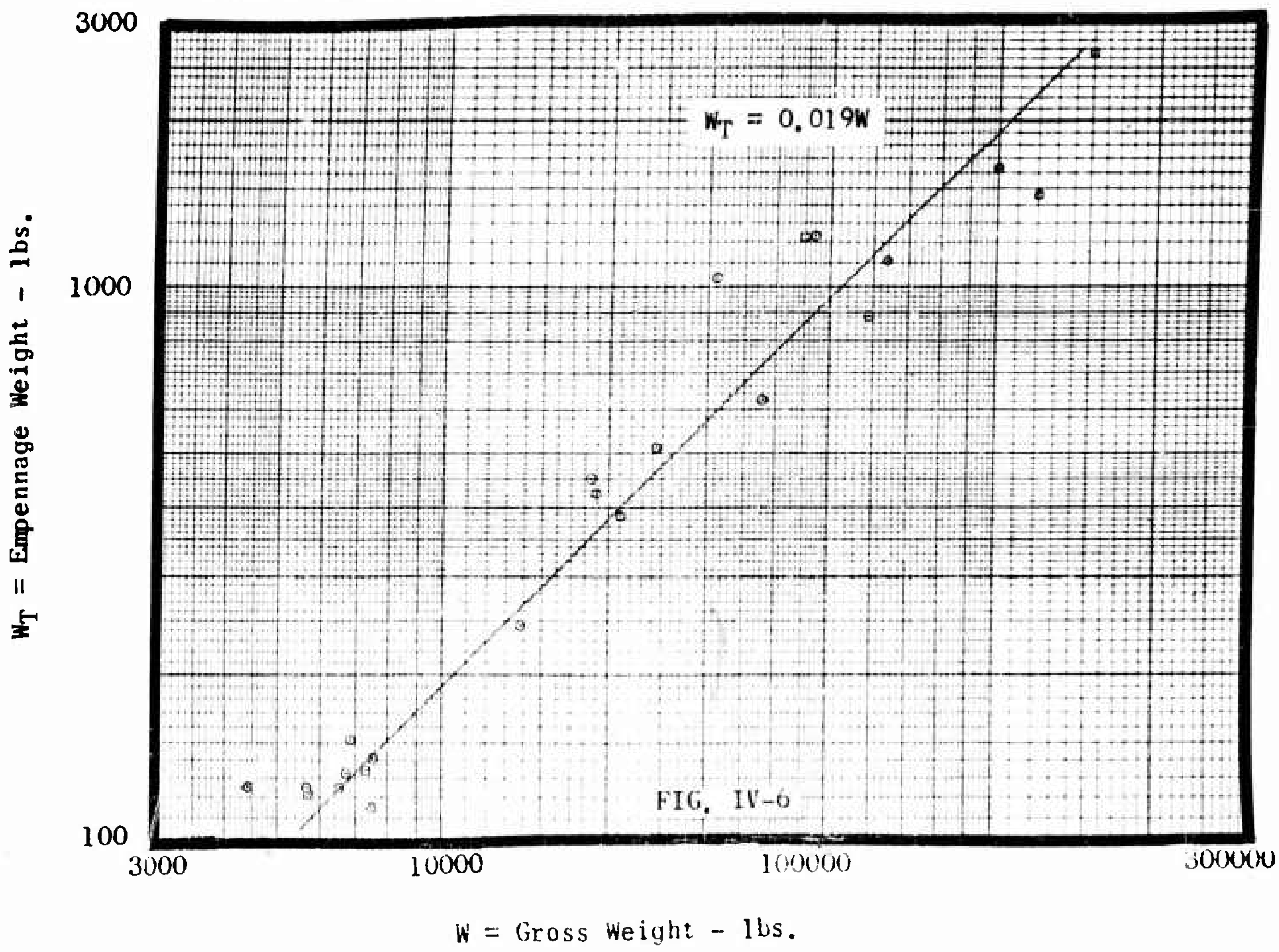


FIG. IV-6

The plot of empennage weight vs. gross weight (Figure IV-6) of many airplanes reveals that empennage weight for airplanes may be reasonably expressed as 1.9% of gross weight.

3. Application of Trends

Tandem Helicopters

Average tail of 1% of gross weight has been used without adjustment.

Compound Single Rotor Helicopter

2.9% of gross weight has been estimated for the shaft driven rotor versions of this aircraft to reflect an anti-torque rotor in addition to stabilizing surfaces that must be capable of withstanding the forward speeds associated with this concept.

Utilization of a tip rocket driven rotor obviously eliminates the need for an anti-torque device, hence an allowance of 1.9% gross weight has been made for the tail group weight for this configuration assuming that its tail design would closely approach that of a fixed wing aircraft.

Retractoplane

The retractoplane in forward flight is fundamentally a fixed wing aircraft, therefore its tail group weight has been assumed to consist of 1.9% gross weight for its stabilizing surfaces plus an additional 1.1% gross weight for an anti-torque rotor giving a total of 3.0% gross weight as the total tail group weight for the shaft driven rotor design. In the case of a tip rocket driven rotor no anti-torque device is necessary, therefore, only 1.9% gross weight has been used.

Fixed Wing VTOL Concept

It has been assumed that for every concept of a fixed wing VTOL under consideration some form of positive control force is needed in the tail during vertical and transition flight. In addition to this, a conventional fixed wing type of empennage is required. In order to meet these requirements 3% of gross weight has been estimated for the tail group weight, assuming that the necessary control force can be supplied by control units weighing about 1.1% gross weight.

F. BODY GROUP

1. Body Group Weight Trend

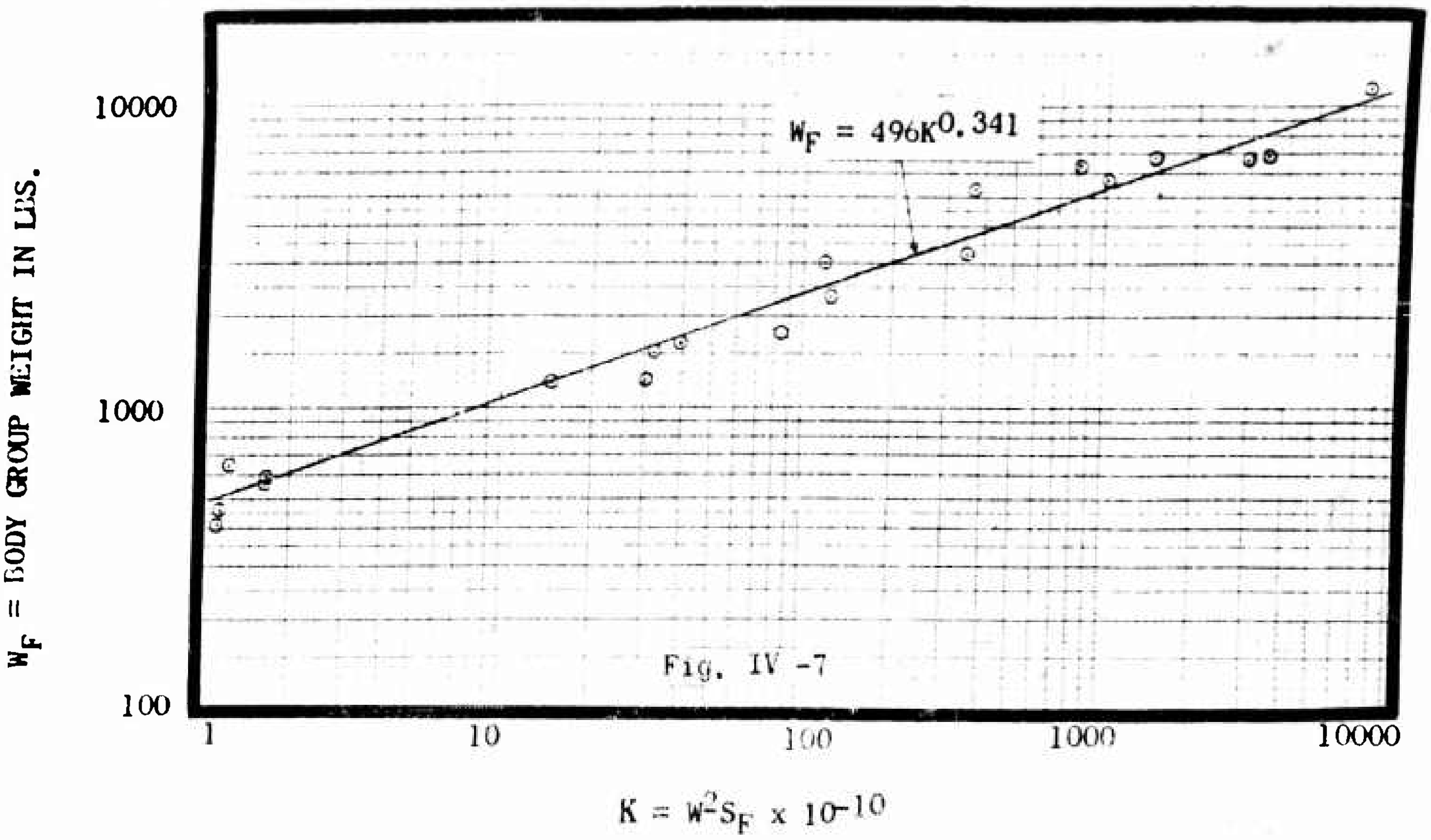


Fig. IV -7

The body group weights of various fixed wing aircraft plotted against gross weight squared times fuselage surface area (Figure IV-7) yields an acceptable trend. The scatter of points should be expected considering the wide variation in purpose, design parameters, and construction details of the airplanes represented. This trend has been used as the basis for body group weights for each VTOL concept presently being scrutinized, with necessary adjustments made to satisfy individual special requirements.

2. Application of Trends

Helicopter

The fixed wing body group trend has been increased by 70% to allow for the fore and aft body loads associated with a tandem rotor fuselage. The resultant weight is approximately 1% gross weight lighter than present day helicopters, which are much smaller than an aircraft which could perform the mission specified for this study. It is reasonable to assume that the body group weight will become a smaller percentage of gross weight as size and gross weight increases.

A single rotor helicopter fuselage more closely approaches that of a fixed wing aircraft, however, the rotor transmission and its associated torque must be contended with. Therefore, the trend of Figure IV-7 has been increased by 26%. In the case of a tip rocket driven rotor an increase of only 15% was added to the basic trend.

To allow for the requirements of retracting the rotor inside the skin line of the fuselage for the retractoplane versions, the trend weight has been increased by 33% for the shaft driven rotors and 21% for the tip rocket driven rotors.

Fixed Wing Version

The trend curve has been used directly for all fixed wing versions for they all have basically a conventional airplane type fuselage.

G. ALIGHTING GEAR GROUP

1. Alighting Gear Weight Trends

Present day non-retracting helicopter alighting gear weighs an average of 3.5% gross weight. It is estimated that a retractable helicopter type gear can be built for an additional 0.5% gross weight giving a total of 4% gross weight.

2. Application of Trends

Assuming that these VTOL aircraft will be landing as a helicopter, the abovementioned 4% of gross weight represents alighting gear weight for every concept, except the tilting wing turboprop and the vectored lift versions. An allowance of 4.5% gross weight has been used for the tilt wing-propeller version because the gear probably will not be installed on the wing making a longer than normal alighting gear necessary. The vectored slipstream version requires a rather steep angle between the fuselage and the ground, necessitating an unusually high alighting gear. An allowance of 5% gross weight was made to fill this requirement.

H. PROPULSION GROUP

The propulsion group includes the following items: engine section; engine; engine accessories; power plant controls; starting system; and oil cooling system. In the following discussion these items will be referred to collectively as "installed weight." The application of the various propulsion group weights may be seen in the summary charts on pages 34 through 37.

Gas Turbine - No Propeller Reduction Gearing

Present day installed weight averages 0.56 lbs./HP. The engines alone weigh 0.33 lbs./HP leaving 0.23 lbs./HP for the remaining installation items. It is predicted that, by 1962, these engines will weigh 0.24 lbs./HP Assuming engine size for a given horsepower will decrease and the development of more efficient starting systems, the installation weight has been reduced in the same proportion as the engine weight giving a total installed weight of 0.42 lbs./HP.

Gas Turbine - Propeller Reduction Gearing

It is estimated that by 1962 a turboprop engine will weigh approximately .33 lbs./HP. Assuming the same installation weights as used above, the total installed weight will be 0.51 lbs./HP. For the Breguet-Kappus concept the installed engine weight has been increased by 10%, to allow for the ducting and valves necessary to channel the exhaust from the gas producer either to the propeller turbines, or straight aft for forward propulsion. This results in an installed weight of 0.56 lbs./HP.

The tilting ducted propeller and the Aerodyne are assumed to have 3 or more adjacent engines. Therefore, the engine weight has been increased to .53 lbs./HP. to allow for a common mixing gear box.

Turbo-jet

A non-augmented turbo-jet engine has been assumed to weigh 0.208 lbs./lb. thrust by 1962. The installation weight has been estimated to be 1/3 of the engine weight giving a total installed weight of .277 lbs./lb. thrust.

For the deflected jet thrust concept, an additional .013 lbs./lb. thrust has been estimated for the thrust deflecting mechanism. This gives a total of 0.29 lbs./lb. thrust for total installed weight.

Turbine Rockets

Based on information from Ref. 1 the installed weight of turbine rockets, including fuel system, has been assumed to weigh .09 lbs./HP plus 360 lbs. per engine.

Rotor Tip Rocket

Based on Ref. 1 rotor tip rocket has been assumed to weigh 0.145 lbs./lb. thrust, including the fuel system.

Low Specific Weight - Short Life Turbo-jets

It has been assumed that engines designed for vertical flight and hovering only and installed in clusters could be built for approximately 0.09 lbs./lb. thrust for the engine alone plus an additional one third of the engine weight for installation, yielding a total installed weight of 0.12 lbs./lb. thrust.

By-Pass Turbojet

This engine, commonly referred to as a "ducted fan", is estimated to weigh 0.23 lbs. per lb. thrust. The installed weight is assumed to be 0.306 lb. per lbs. thrust, allowing 33% for installation weight. For the deflected thrust concept this weight has been increased to 0.345 lbs. per lb. thrust to include the weight for a deflection device.

I. DRIVE SYSTEM

1. Rotor Drive System Weight Trend

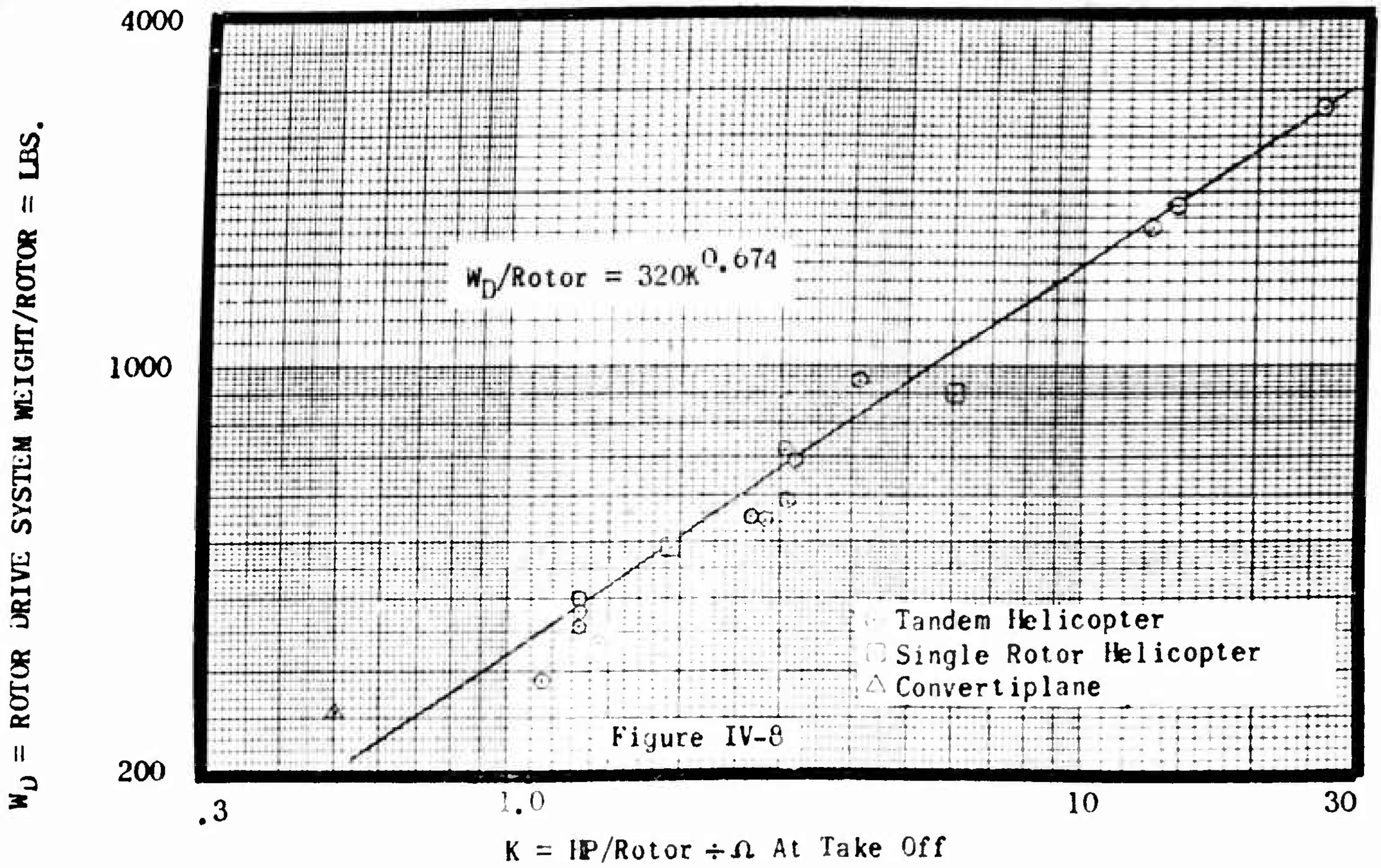


Figure IV-8

For simplicity and ease of calculation the entire rotor drive system weight per rotor has been correlated with take-off horsepower and rotor RPM, in lieu of investigating the various component transmissions and shafting separately. By using weight and design horsepower per rotor, both single and tandem rotor helicopters can be included in the same trend expression. For tandem helicopters it has been assumed that each rotor is designed to absorb 60% of the total available horsepower. Although both reciprocating engine and gas turbine powered systems are plotted in Figure IV-8 the trend has been determined primarily by gas turbine systems, because reciprocating engines will not be used in any of the VTOL concepts. The drive system includes all components necessary to reduce the turbine speed down to the rotor RPM including the rotor drive shaft. It does not include the anti-torque transmissions or shafting which are included, when necessary, in the tail group weight.

2. Propeller Synchronizing Shafting and Transmissions Weight Trend

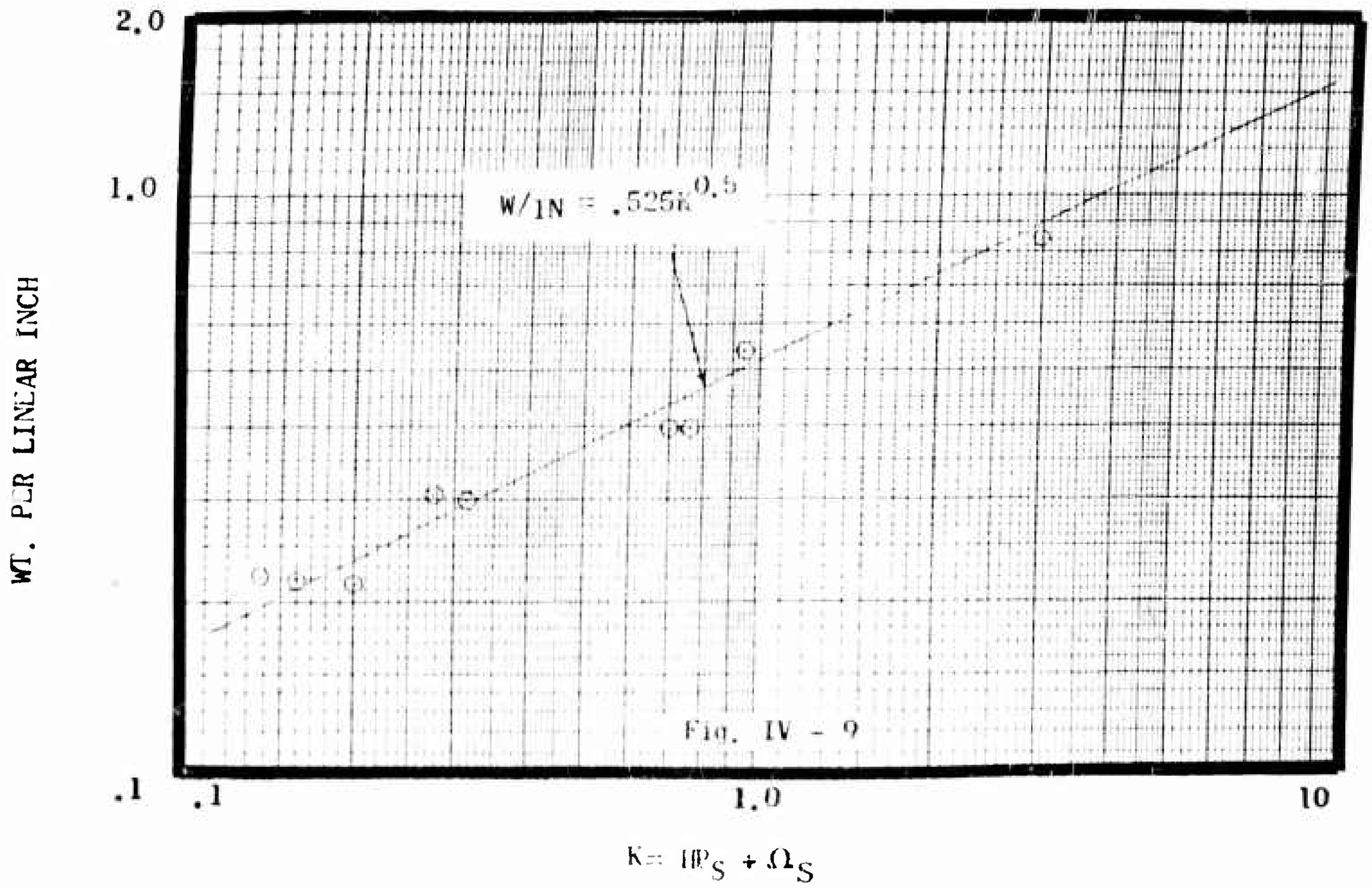


Fig. IV - 9

In Figure IV-9 the weight of rotor synchronizing shafting, couplings and bearings from tandem helicopters have been related to the torque transmitted, torque being represented by horsepower divided by shaft RPM. It has been assumed that propeller synchronizing shafting, when needed, will follow this trend.

To estimate the gearing necessary to connect this shafting to the propellers, a line having the same slope as the synchronizing shafting trend was drawn through the plot of three tandem helicopter intermediate transmission weights vs. horsepower divided by RPM. To better adapt this trend for use in this study it has been reduced by 25% to reflect a two gear transmission instead of three gears as contained in the intermediate transmissions investigated. The resulting expression is:

$W_{SX} = 130\ (HP_X \div \Omega_X)^{0.5}$ x No. Transmissions

3. Application of Trends

Rotary Wing Concepts

A 4.5% reduction in the present trend weight of the rotor drive system has been made to reflect the "state of the art" in 1962. With this reduction, the equation of the drive system becomes, for tandem helicopters:

$$W_D = (305 \times 2)(0.6\ HP_{TO} \div \Omega_{TO})^{0.674} \quad \text{and}$$

for single rotor compound helicopters:

$$W_D = (305)(HP_{TO} \div \Omega_{TO})^{0.674}$$

Furthermore, for the retractoplane, the rotor drive weight trend has been reduced by approximately 12% to reflect the limited life system utilized by this aircraft. Accordingly, the expression for the retractoplane system becomes:

$$W_D = 270\ (HP_{TO} \div \Omega_{TO})^{0.674}$$

The compound helicopter and retractoplane concepts powered by gas turbines installed in the wing will require synchronizing shafting and transmissions. Therefore the weight (Figure IV-9) of these items must be added to the rotor drive weight.

The drive system of the tip rocket driven rotors only consists of a rotor shaft and bearings. An estimate of 0.1W has been based on a review of rotor drive shaft weights of present day helicopters, taking account for the absence of driving torque but remembering that the shaft must still transmit lift and control forces of the helicopter.

Fixed Wing Concepts

Propeller synchronization is contemplated in the turboprop tilting wing, tilting ducted propeller, vectored lift, Breguet Kappus, and Aerodyne VTOL concepts. To account for this weight the trends of Figure IV-9 have been used. In addition to this, a weight allowance for propeller extension shafts is represented by the expression:

$$W_{PS} = 7.2\ (HP_P \div \Omega_P)(L)$$

This expression reflects the weight of helicopter rotor drive shafts, which have been designed by the rotor torque and bending moments imposed by various maneuvers, similar to the conditions that would be imposed on a VTOL propeller extension shaft. In retrospect, this expression yields a weight that appears to be conservative (heavy) but is believed to be within the accuracy of this study.

J. FIXED EQUIPMENT

1. Weight Trend

A minimum of fixed equipment has been estimated to meet the requirements of the transport VTOL mission. Fixed equipment weight excluding flight controls has been considered constant for all the configurations. A list of the main items and their assumed weight follows:

Instruments	230 lbs.
Hydraulic System	350 lbs.
Electrical System	850 lbs.
Communicating	300 lbs.
Furnishing (no de-icing)	650 lbs.
TOTAL CONSTANT WEIGHT	2380

Flight or surface controls for airplanes within the anticipated gross weight range of the study average approximately 1% gross. A reasonable estimate for helicopter controls, including a hydraulic boost system, is 3% gross weight. Flight control weight trends for each concept will be based upon these two systems.

2. Application of Trends

Tandem Helicopter

Conventional controls equal 3% W.

Compound Helicopter

Combination of airplane and helicopter controls are needed. Assuming that some portions of the system need not be duplicated, the weight is estimated to be 3.5% W.

Retractoplane

A combination of airplane and helicopter controls plus rotor retracting controls is estimated to weigh 4.5% W.

Tilting Wing Turbojet and By-Pass Turbine, Deflected Turbojet and By-Pass Turbine, Vectored Lift Turboprop, and Direct Jet Lift

Conventional airplane controls plus 0.5% W for special control problems giving a total flight control weight of 1.5% W.

Tilting Ducted Propeller, Breguet-Kappus, and Aerodyne

Conventional airplane surface controls plus collective pitch controls for the propellers, combine to give a weight estimate of 2.5% W.

Tilting Wing-Turboprop

The weight of this control system is assumed to be the same as the Breguet-Kappus with an additional 0.1% W to allow for controlling the tilt of the wing.

K. ENGINE LUBRICATING SYSTEM

In order to simplify the trend calculations the engine lubricating system has been assumed to weigh 500 lbs. for all engines except tip rockets. Although this weight would in fact vary, the error introduced by this assumption is relatively small and will have a negligible effect on the trends.

L. USEFUL LOAD LESS FUEL

All useful load items, except for fuel and trapped liquids have been assumed to remain constant. The trapped liquid weight has been varied to reflect the difference between reduction gear box oil requirements for various configurations. The useful load less fuel breakdown is shown in Figure IV - 10.

	USEFUL LOAD LESS FUEL			
	Rotary Wing VTOL		Fixed Wing VTOL	
Item	Shaft Driven Rotor	Tip Rocket Driven Rotor	Turboprop Types	Turbo-Jet Ducted Fan Etc.
Crew (3)	600	600	600	600
Trapped Liquids (Trapped Fuel & Oil, Reduction Gear Box Oil & Engine Oil)	850	400	650	500
Payload	8000	8000	8000	8000
Miscellaneous	50	50	50	50
USEFUL LOAD LESS FUEL	9500	9050	9300	9150

FIG. IV - 10 - USEFUL LOAD LESS FUEL

M. FUEL AND FUEL SYSTEM

For gas turbine and turbojet engines, JP-4 fuel weighing 6.5 pounds per gallon has been used. Fuel systems for these engines are assumed to weigh 0.2 lbs. per gallon for fuselage mounted tanks and 0.15 lbs. per gallon for wing tanks.

Fuel weight for the tip rockets has been computed from the fuel consumption of 15 pounds per pound thrust per hour (see Figure V - 13) Similarly, fuel weight for the turbine rocket engines has been based on a fuel consumption of 8½ pounds per shaft horsepower per hour (see Figure V - 17) The fuel system weight for both these systems has been included in their propulsion group weight.

V. POWER PLANT TREND DATA *

A. INTRODUCTION

Since the performance and therefore the competitive position of VTOL aircraft is dependent to a large extent on low specific fuel consumption and low specific weight power plants, the need to predict future power plant design trends accurately is exceedingly important for this type of study. Power plant performance representative of the state of the art of the year 1962 is used for the VTOL transport study. Figure V-1 presents predicted 1962 state of the art performance and weight data for various engine types considered for this study. Performance and weight data for various current, development and study engines has been obtained from cognizant engine manufacturers and is listed in Figure V-2. The specific fuel consumption and specific weight of representative shaft turbine, turbojet and ducted fan engines is plotted against the date of availability to allow the construction of curves representing the trend of technological improvement from which predicted 1962 values are obtained. The reduction of available power for operation at a pressure altitude of 6000 feet and at an ambient temperature of 95 degrees F. represents an average of the estimates obtained from representative engine manufacturers. Figures V-5, V-8 and V-11 depict the average ratios of part load specific fuel consumption at various flight speeds to static military rating for a typical shaft turbine, turbojet and ducted fan engine respectively. Predicted rocket-on-rotor and propellant turbine power plant performance estimates were obtained from Reaction Motors (Ref. 1). The reciprocating engine is not considered as a candidate power plant for this study due to its bulk, installed weight and development stagnation.

FIGURE V-I

	Shaft Turbine Geared approx. 1000 RPM	Shaft Turbine Direct approx. 10500 RPM	Turbojet over 10000# Thrust after-burner	Turbojet over 10000# Thrust non-augmented	2450# Thrust non-augmented	By-Pass Turbo-Jet	Rocket on Rotor	Propellant Shaft Turbine
Specific Weight (#/SHP Mil.)	.327	.238						.09 + 360# per engine
Specific Weight (#/ESHP Mil.)	.309	.226						
Specific Weight (#/# Thrust TO)			.174	.208	.10			
Specific Weight (#/# Thrust Mil.)			.264	.208	.10	.230	.145	
Specific Fuel Con. (#/SHP Mil/hr.)	.500	.500						8.5
Specific Fuel Con. (#/ESHP Mil/hr.)	.474	.474						
Specific Fuel Con. (#/# Thrust TO/hr)			1.50	.798	.91			
Specific Fuel Con. (#/# Thrust Mil/hr)			.82	.798	.91	.590	15.0	
% Power Available at 6000' at 95°F	67%	67%	72%	72%	72%	70%	103%	103%
Part Load Fuel Consumption — See Fig.	V-5	V-5	V-8	V-8	V-8	V-11	V-14	V-18

* Prepared by: W. Godon

FIGURE V-2

ENGINE SUMMARY LIST

Manufacturer	Type	Model No.	Specification or Report Number	Take-off SHP	Military SHP	Engine Weight Pounds	Military SFC	Date of Availability		
Allison	Shaft Turbine	XT38-A-3	374-A	1000	1000	1240	.800	1954		
		T40-A-6	300-D----	5302	5302	2064	.603	Jan 1954	•	
		YT56-A-1	276-F	3017	3017	1575	.616	Jan 1955	•	
		YT56-A-5	391	3490	3490	1120	.580	Oct 1956	•	a
		T56-A-1	339-B	3460	3460	1645	.585	Jan 1955	•	
		501-D8	377-B	3755	3755	1660	.561	Mar 1957	•	a
		500-C14	302	7510	7510	3150	.560	36 MFGA	•	
		500-C15	303	6920	6920	3300	.585	48 MFGA		
Allison		550-B1	394-A	5200	5200	2150	.508	Sept1959		
Proposed		Twin Spool	Proposed	5930	5930	2150	.524	Sept1959	•	
General Electric		XT50-GE-2	SE-1	1024	1024	325	.660	Summer 1956	•	a
Pratt & Whitney		T-34	3529	5500	5300	2590	.695	1953		
Pratt & Whitney		T-57		13340	13340	6600	.606	Dec 1958	•	
Westinghouse		RB109	TSD449	4020	3200	1850	.515	1958	•	
Lycoming		XT55-L-1	127.1	1595	1458	600	.707	Dec 1957	•	a
Curtiss-Wright	Shaft Turbine	YT49-W-1	875-E	8500	8500	4466	.803	Current		
				(# Thrust)	(#Thrust)					
General Electric	Ducted Fan	X84	R54AGT105	32900	16900	5100	.619	unknown		
		X84A	R55AGT22	17400	17400	4300	.593	Jul 1959	•	
Curtiss-Wright		WTF4	AC-215A	32000	18000	7000	.640	1960		
		WTF5	AC-216A	19200	19200	5500	.605	1960	•	
Westinghouse		R.Co.7	TSD 568	13600	12002	3731	.722	1957	•	
	Ducted	PD42-1		16500	16500	3550	.690	1961	•	
	Fan	PD42-2	WAGT F42.2.1	27200	16000	5425	.715	1962		
Allison	Turbojet	J71-A-2	361-C	14000	9850	4889	.955	Jul 1955		
		J71-A-9	356-B	9570	95.0	4090	.880	May 1954		
		J71-A-11	381-B	9700	9700	4090	.880	Apr 1955		b
		600-B44	403	13600	9500	4890	.900	Apr 1957		b
		700-PD8	0000-WPD-X12	18000	12000	3280	.825	unknown	•	
		700-PD9	0000-HPD-X12	37500	25000	7320	.825	unknown	•	
General Electric		J47-GE-15	E-582	6000	5200	2515	1.130	1949	•	
		J47-GE-23	E-591-B	5910	5620	2512	1.028	1951	•	
		J73-GE-3			8920	3880	.917	1952	•	
		J79-3	R53AGT78	14350	9300	3255	.860	Sept1956		
		J79-216	R55AFT400	15600	10000	3255	.839	Jul 1957	•	b
		J79-207	R54AGT571	18000	12000	3500	.834	Jul 1959	•	
		MX2273	R55SE5	2450	2450	231.4	.910	Spring 1957		
		SJ-110-C1	R55SE19	3520	2470	327	.99	Nov 1957		
General Electric		SJ-110-C3	R55SE19	3621	2470	333.1	.99	Nov 1957		
Pratt & Whitney		J57-1	1680	12500	11200	3790	.775	Fall 1956		
		J57-2	1696	13750	11200	3865	.775	Summer 1957		
		J57-20	1681	17200	10950	4720	.810	Apr 1957		
		J75-1	1660	15800	15800	5300	.770	Mar 1957		b
		J75-24	2604	23500	15500	6100	.800	Mar 1957	•	b
		J75-21	5900	25000	16500		.830	Aug 1958		
Pratt & Whitney		J52 A/B	Inst. Hbk.	11000	7250	2750	.820	1960	•	
Pratt & Whitney		J52		7800	7800	2000	.820	1960	•	
Curtiss-Wright		J65-W-4	N890-A	7700	7700	2750	.915	1955		
		J65-W-6	N898	11000	7600	3485	.930	Jul 1955	•	b
		J65-W-7	892-E					1955	•	b
Westinghouse		PD33-1	WAGT220B-C	6075	6075	1425	.860	1957	•	
		PD33-2	WAGT120B-B	10000	6800	1960	.950	1958	•	
Fairchild		FT100A	298	2450	2450	325	.940	unknown		
		FT100B	301	3550	2360	415	.900	unknown		

B. SHAFT TURBINE ENGINES

1. Specific Weight Trend

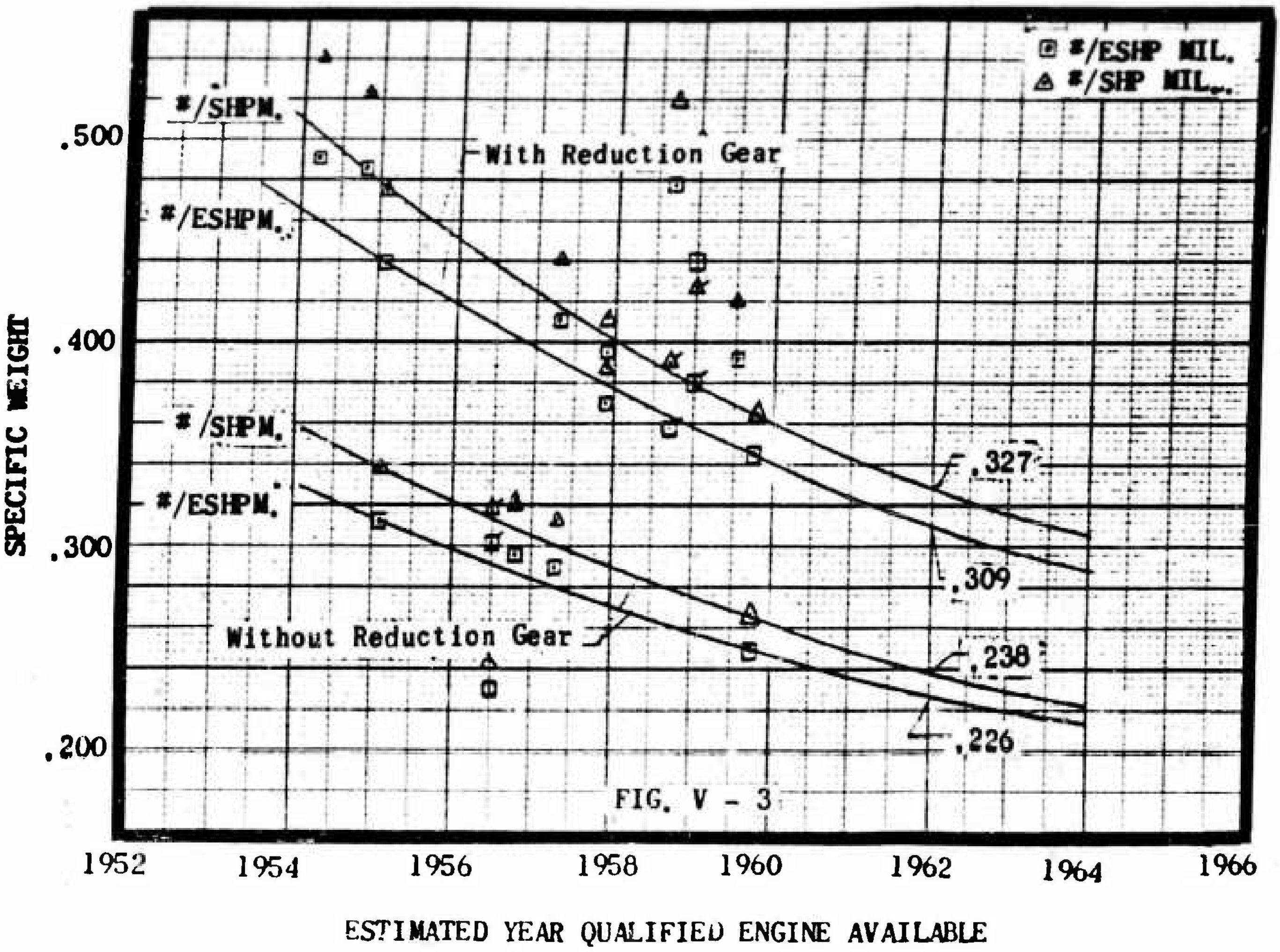


FIG. V - 3

Figure V-3 presents specific weight trend curves for shaft turbine engines with reduction gearing to provide a shaft speed of approximately 1,000 RPM and for direct drive turbines with shaft speeds of approximately 10,500 RPM. The specific weights of typical shaft turbine engines as indicated by an asterisk (*) in Figure V-2 have been plotted against the year in which these engines are scheduled to complete their qualification testing.

2. Specific Fuel Consumption Trend

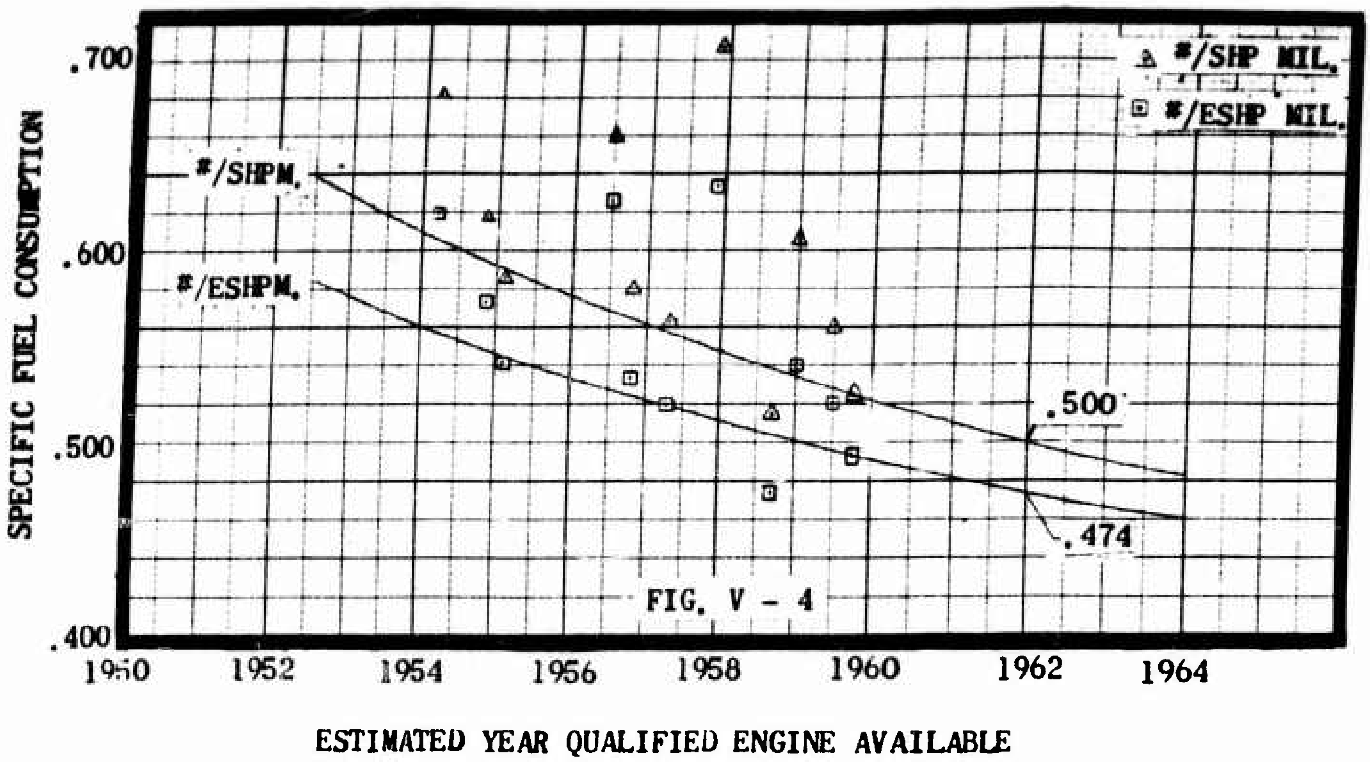


FIG. V - 4

Figure V-4 presents specific fuel consumption trend curves constructed by plotting specific fuel consumption values of typical shaft turbine engines as indicated by an asterisk (*) in Figure V-2 against the year in which these engines are scheduled to complete their qualification testing.

The most advanced shaft turbine engine for which performance and weight data was obtained is the Allison twin spool 550 B1 for which specification number 394-A reports the following preliminary performance based on military power at sea level static conditions:

Dry weight with reduction gear (1,000 RPM)	2,150 lbs.
Dry weight for direct drive engine (10,500 RPM)	1,575 lbs.
Equivalent shaft horsepower	5,500
Shaft horsepower	5,200
Specific weight with reduction gearing (lb./SHP)	.413
Specific weight for direct drive engine (lb./SHP)	.303
Specific fuel consumption (lb./SHP/Hr)	.508
Scheduled date of production availability	Sept. 1959

For the purpose of this study it was assumed that, should a military requirement exist, a twin spool shaft turbine engine similar to the 550 B1 could be developed in the same period of time for the same weight which would realize a 14% power growth at the expense of a 3% increase in specific fuel consumption. The values of specific fuel consumption and specific weight of this proposed engine have been used a control points for the trend curves in Figures V-3 and V-4.

3. Part Load Fuel Consumption Trend

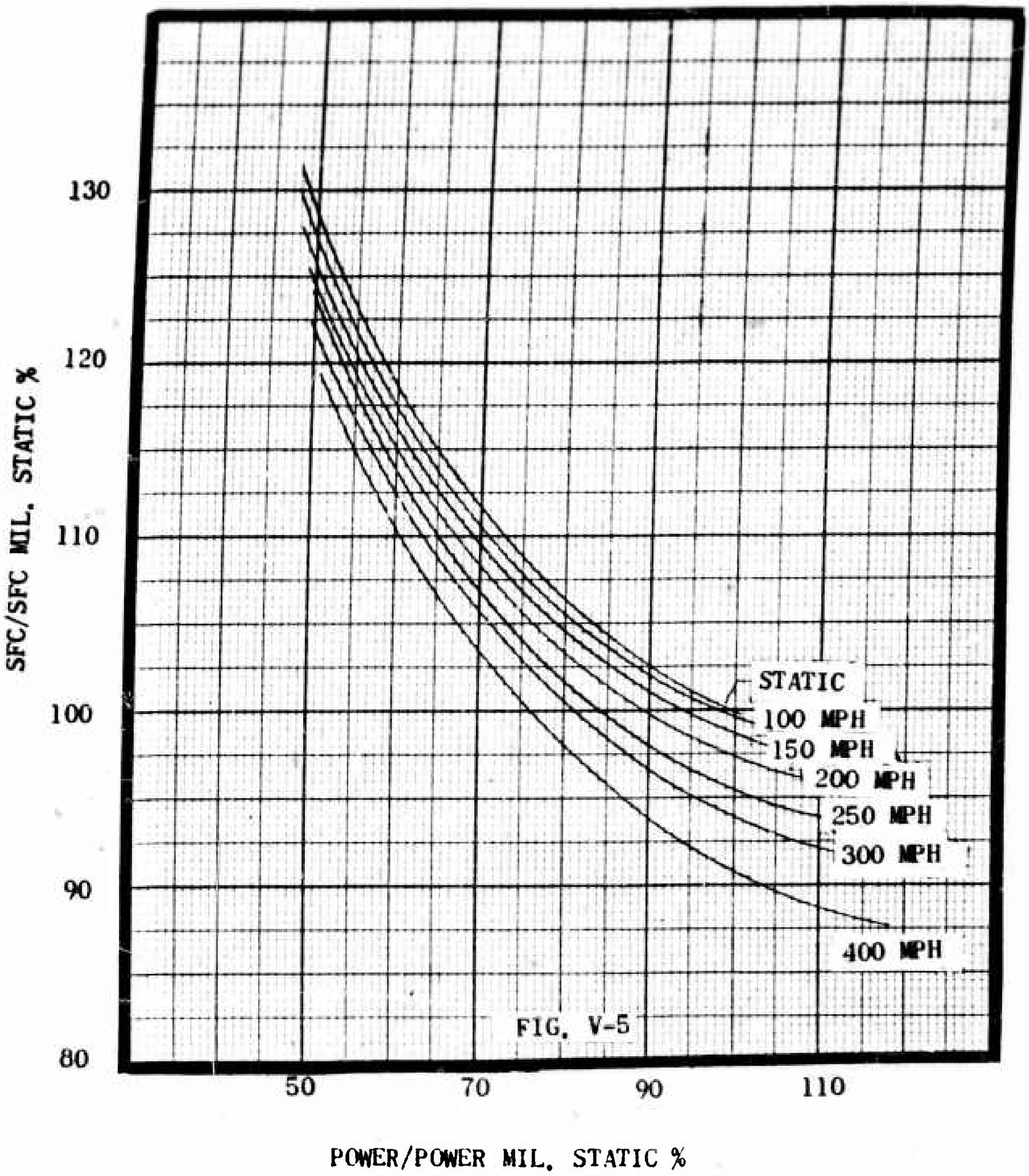


FIG. V-5

Figure V-5 represents the ratio of part load specific fuel consumption at the noted flight speeds to static military power specific fuel consumption as a function of the ratio of part load power at these flight speeds to static military power for a typical shaft turbine engine. These curves were constructed as averages of these ratios for the shaft turbine engines indicated by an "a" in Figure V-2.

4. Power Variation with Altitude & Ambient Temperature

The power available at a pressure altitude of 6,000 feet and an ambient temperature of 95°F of a typical turboprop engine has been selected as 67% of sea level power rating. This selection was based on the following data from engine manufacturers:

66.3% to 66.9%	-- Allison Division of General Motors Corp.
65.7% to 68.2%	-- General Electric Corp.
65% to 70%	-- Lycoming Division of AVCO Mfg. Corp.
67.5%	-- Westinghouse Electric Corp.
60% to 65%	-- Continental Aviation & Engineering Corp.

C. TURBOJET ENGINES

1. Specific Weight Trend

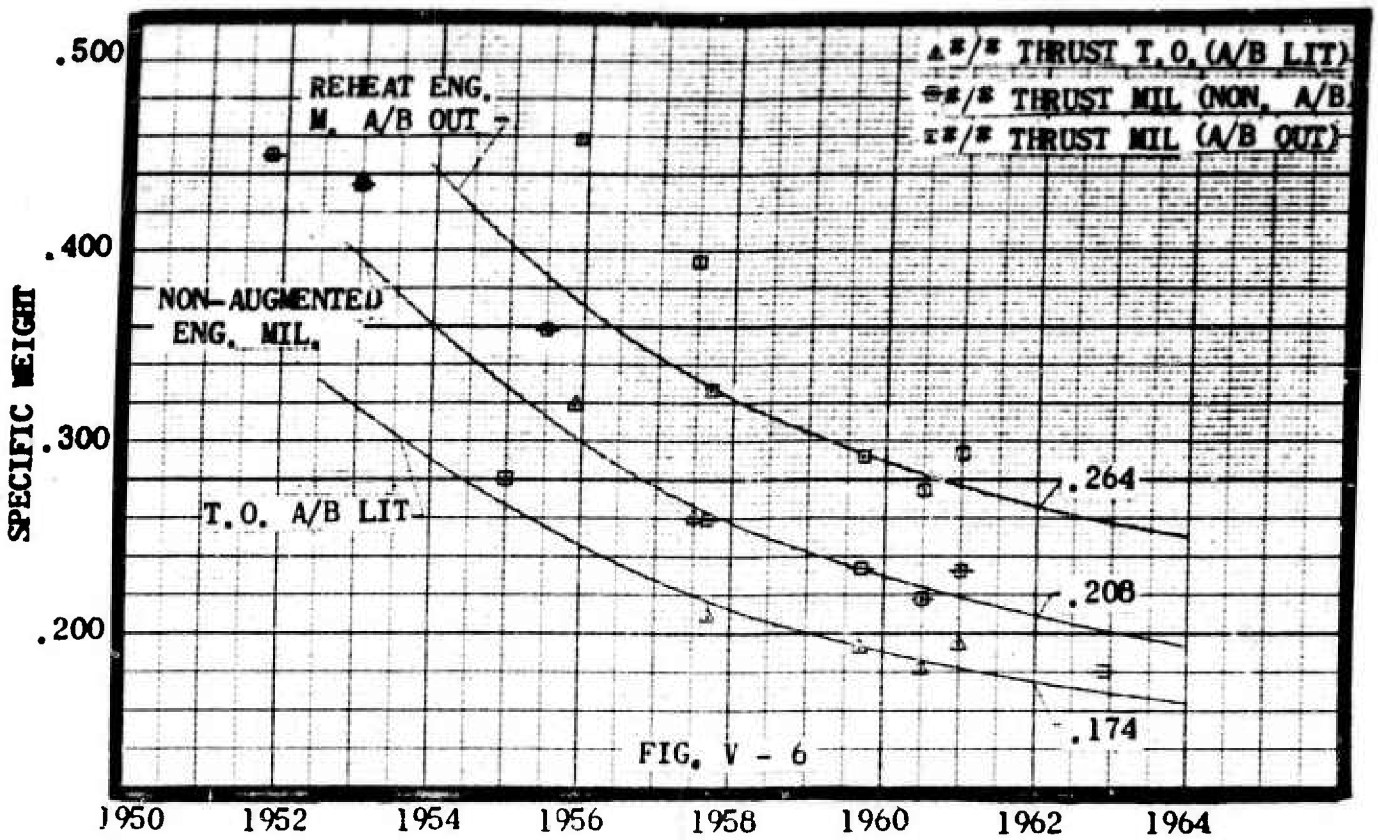


FIG. V - 6

ESTIMATED YEAR QUALIFIED ENGINE AVAILABLE

Figure V-6 presents specific weight trend curves for non-augmented and afterburner equipped turbojet engines. The specific weights of typical turbojet engines as indicated by an asterisk (*) in Figure V-2 have been plotted against the year in which these engines are scheduled to complete their qualification testing.

The values of specific fuel consumption and specific weight of the General Electric J79-X207 single rotor, high pressure ratio turbojet engine, which is scheduled to complete its qualification testing in 1959, have been used as control points for Figures V-6, V-7, and V-8. General Electric has estimated the weight and minimum sea level, static performance of this engine as follows:

Rating	% RPM	Net Thrust Pounds	Net SFC lbs/hr/lb
Maximum	100	18,000	1.670
Military	100	12,000	.834
Cruise	95	9,475	.767
	90	7,500	.745
	85	5,280	.775

Weight with high augmentation afterburner = 3,500 pounds
Weight as a non-augmented engine = 2,800 pounds

2. Specific Fuel Consumption Trend

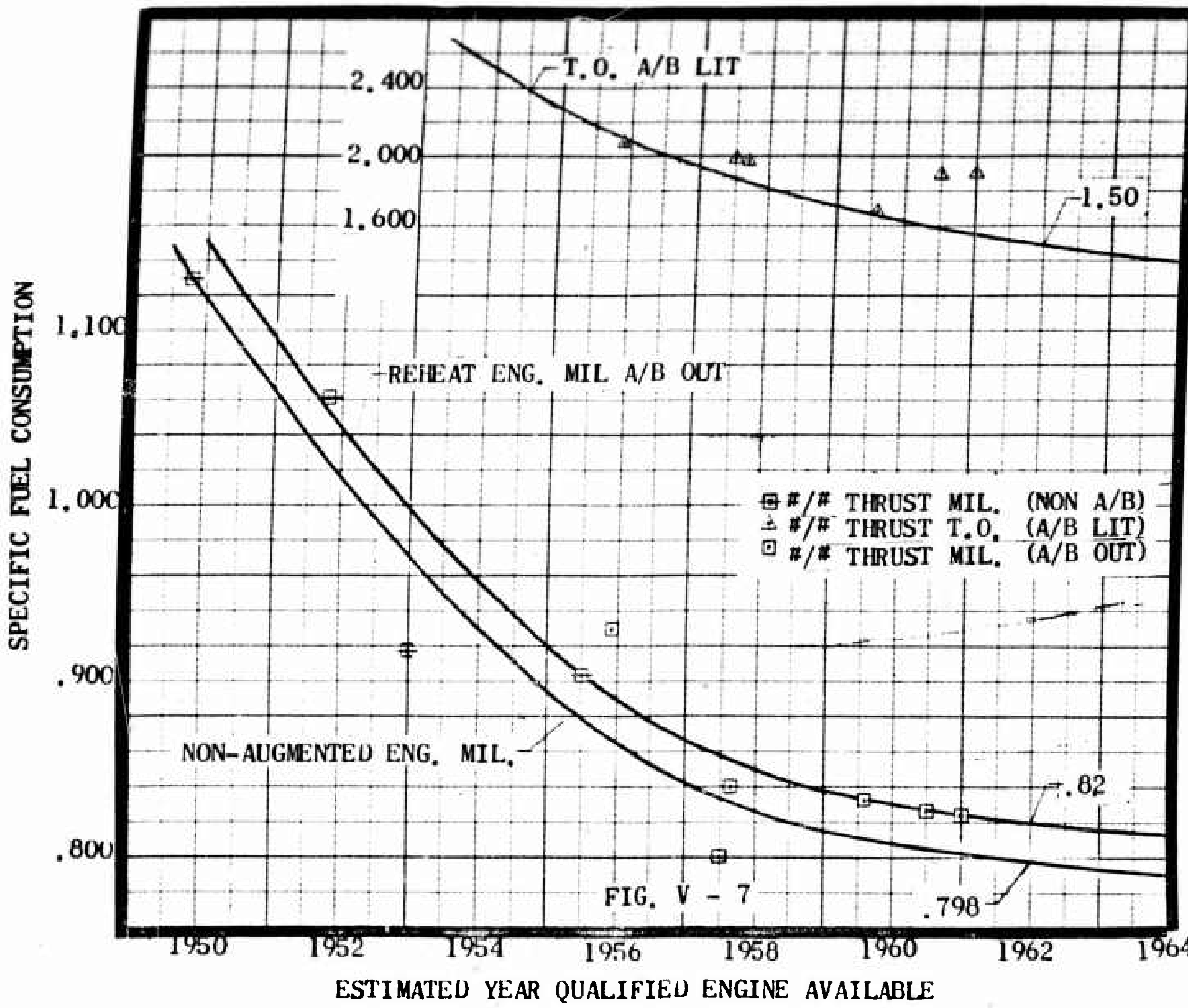


FIG. V - 7

Figure V-7 presents specific fuel consumption trend curves constructed by plotting specific fuel consumption values of typical turbojet engines as indicated by an asterisk (*) in Figure V-2 against the year in which these engines are scheduled to complete their qualification testing. The trend curve for the military power specific fuel consumption of non-augmented engines was determined to be 2.7% lower than the trend curve for the military power specific fuel consumption of reheat engines with the afterburner out. This relationship was determined by averaging the relationship for four engines (indicated by a "b" in Figure V-2) for which both afterburning and non-afterburning model data was available.

3. Part Load Fuel Consumption Trend

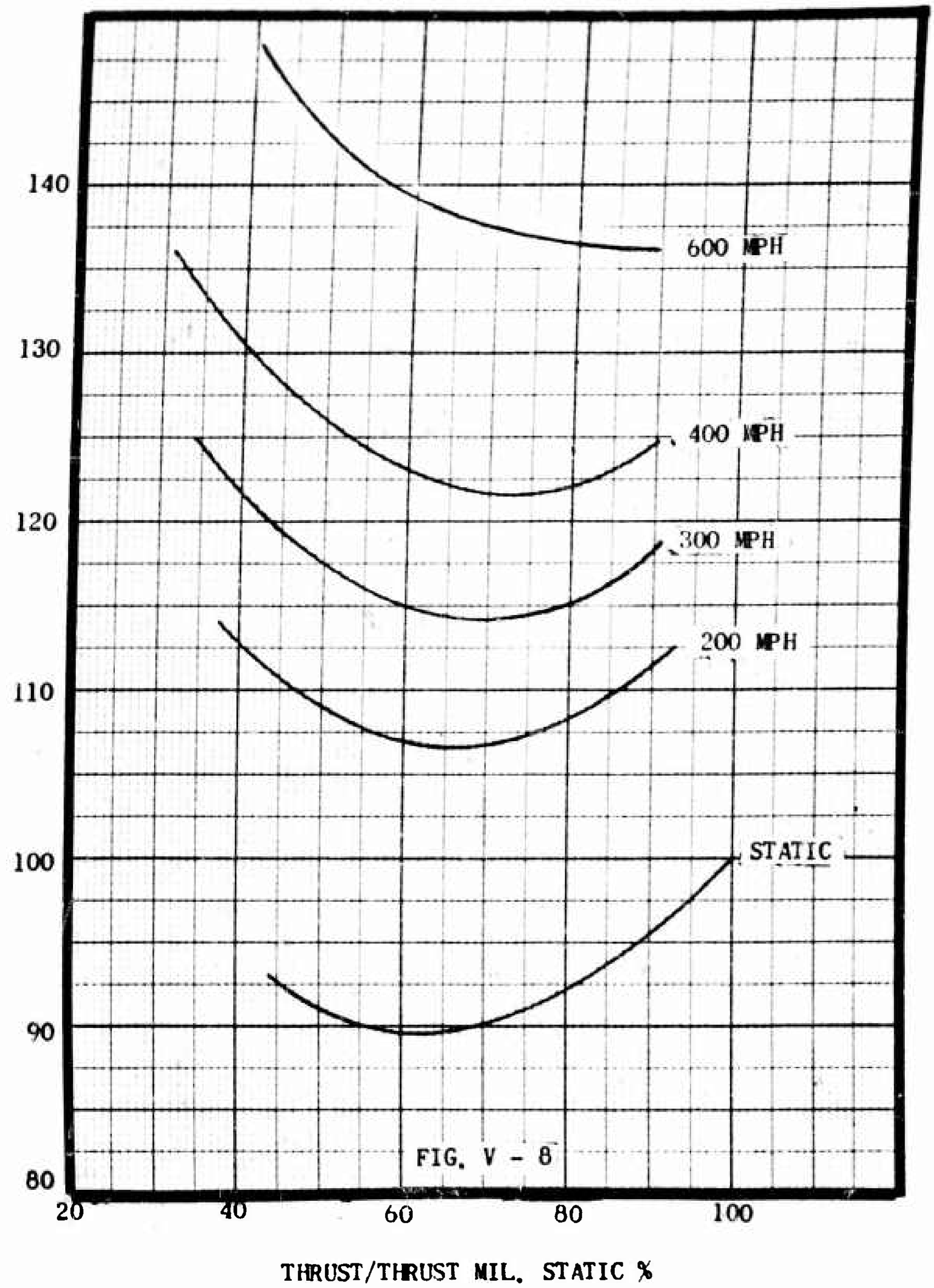


Figure V-8 represents the ratio of part load specific fuel consumption at the noted flight speeds to static military power specific fuel consumption as a function of the ratio of part load thrust at these flight speeds to static military thrust for the General Electric J79-X207 turbojet engine.

4. <u>Thrust Variation with Altitude & Ambient Temperature</u>

The thrust available at a pressure altitude of 6,000 feet and an ambient temperature of 95°F of a typical turbojet engine has been selected as 72% of sea level power rating. This selection was based on the following data from engine manufacturers:

72.5% — Westinghouse Electric Corp.
72% — Continental Aviation & Engineering Corp.
72.5% — General Electric (J79 engine)
72.5% — General Electric (MX2273)

D. DUCTED FAN (BY-PASS TURBOJET) ENGINES

Rand report number R-269 entitled "Performance, Weight, and Size Relations for a Family of Turbofan Engines with Exhaust Mixing" proposes the following characteristics for three non-afterburning, mixed-exhaust turbofan engines based on representative values attainable by engines which were in the development stage during early 1954:

Rand R-269 figure number		89	93	97
Mach No. = 0.8 at	design by-pass ratio	0.75	0.4	0.6
35,332 ft. and normal power	design pressure ratio	12.0	16.0	16.0
Altitude		S.L.	S.L.	S.L.
Flight speed		static	static	static
Weight = pounds		3220	3087	3129
Military thrust = pounds		10000	10000	10000
Specific weight = pounds/pound military thrust		.322	.309	.313
Specific fuel consumption = pounds/pound military thrust/hr.		.458	.616	.512

1. Specific Weight Trend

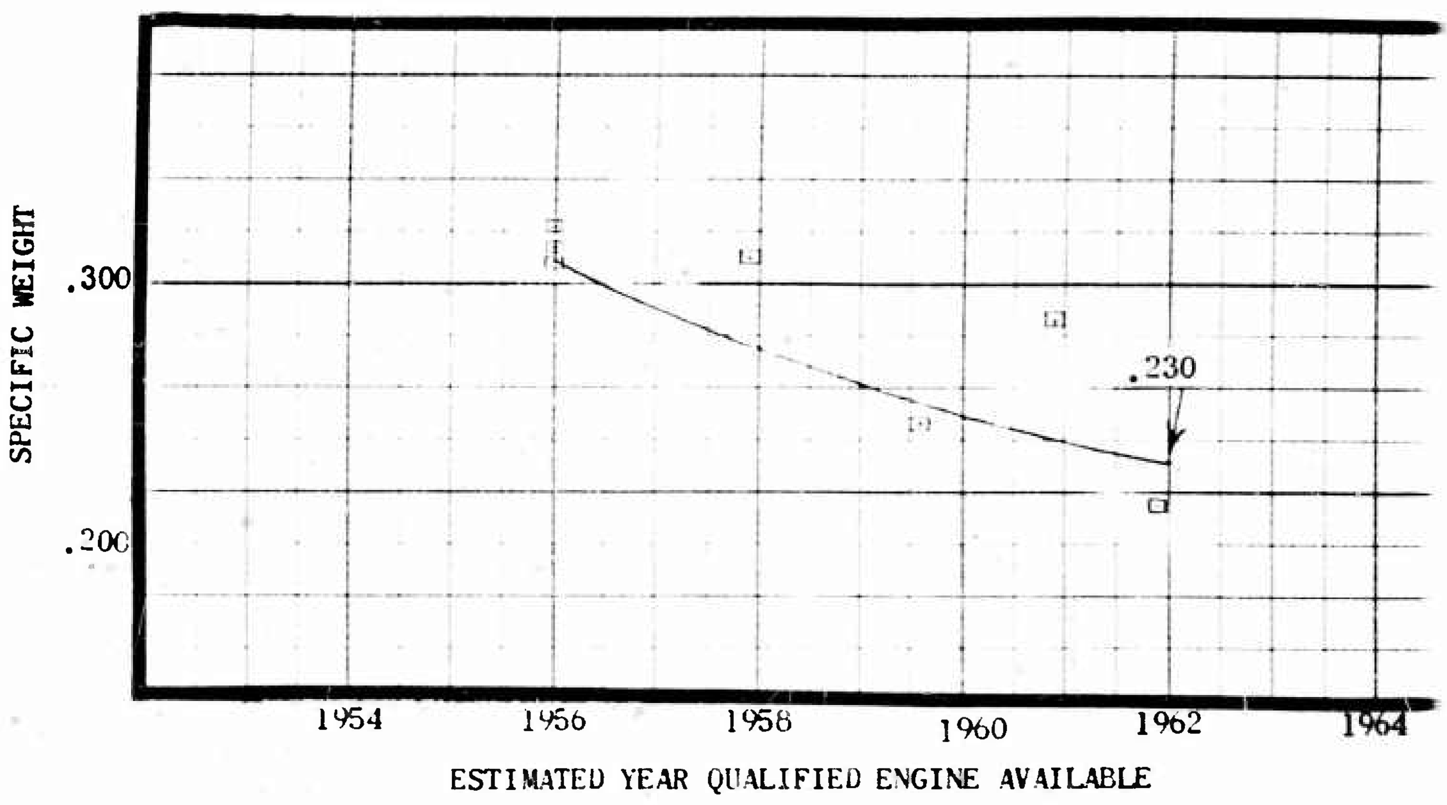


Figure V-9 presents a specific weight trend curve for non-augmented ducted fan engines. The specific weights of typical ducted fan engines as indicated by an asterisk (*) in Figure V-2 and for the three proposed engines from the Rand report have been plotted against the year in which these engines should complete their qualification testing.

2. Specific Fuel Consumption Trend

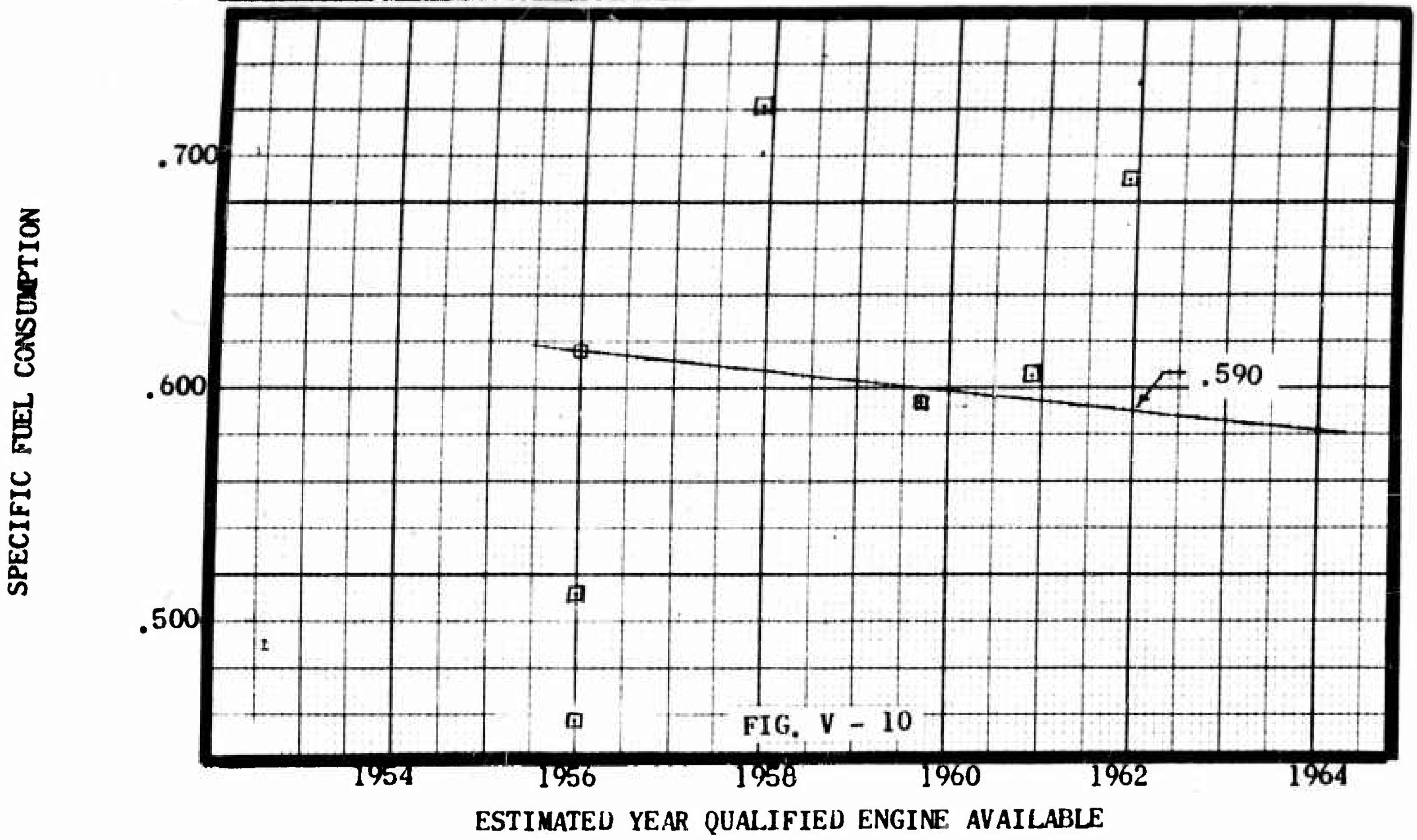


FIG. V - 10

Figure V-10 presents a specific fuel consumption trend curve for non-augmented ducted fan engines. Specific fuel consumption values for typical ducted fan engines as indicated by an asterisk (*) in Figure V-2 and for the three proposed engines from the Rand report have been plotted against the year in which these engines should complete their qualification testing.

3. Part Load Fuel Consumption Trend

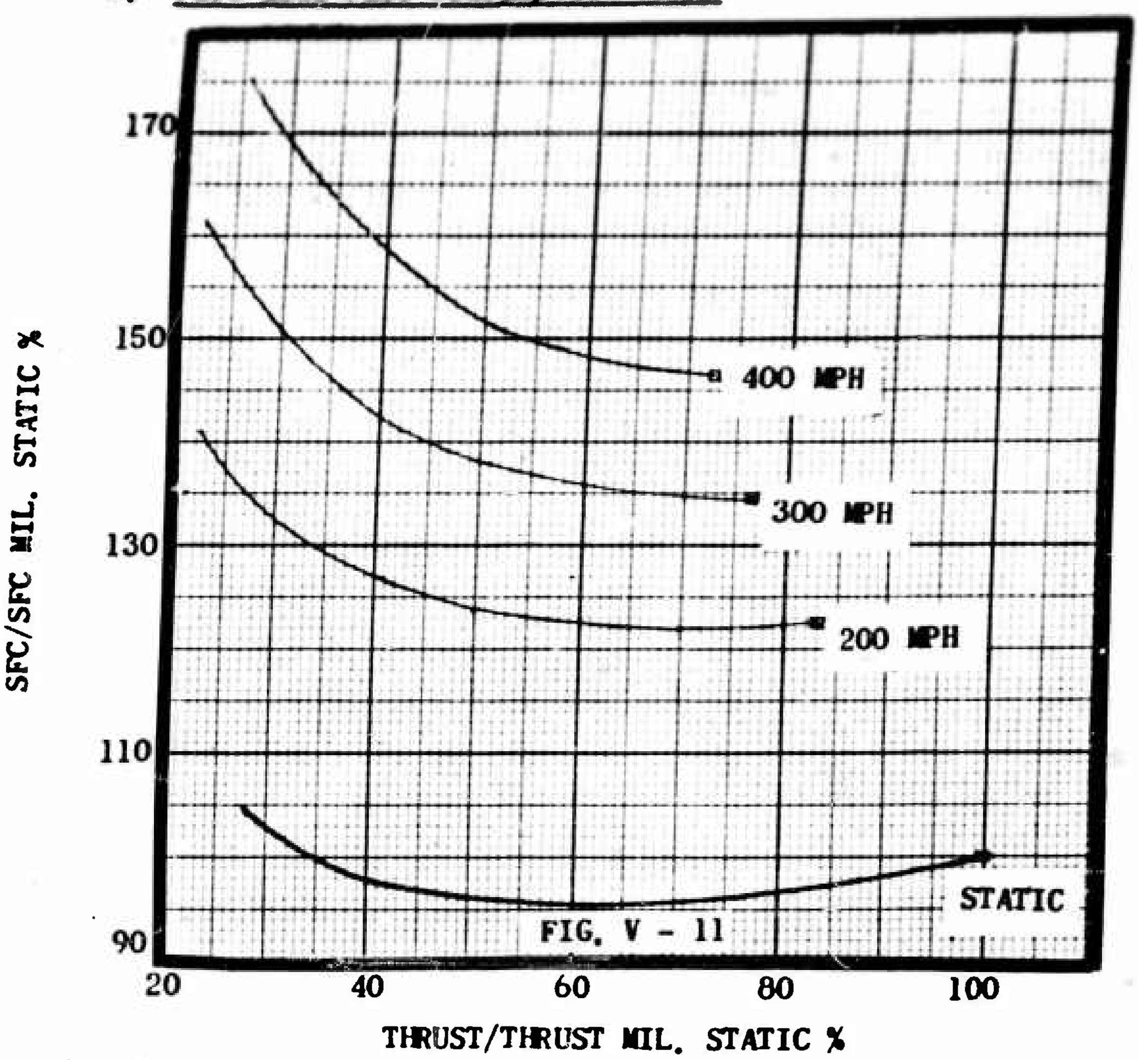


FIG. V - 11

Figure V-11 represents the ratio of part load specific fuel consumption at the noted flight speeds to static military thrust specific fuel consumption as a function of the ratio of part load thrust at these flight speeds to static military thrust for a typical ducted fan engine.

4. **Thrust Variation with Altitude & Ambient Temperature**

The thrust available at a pressure altitude of 6,000 feet and an ambient temperature of 95°F of a typical ducted fan engine has been selected as 70% of sea level thrust rating. This selection was based on the following data from engine manufacturers:

70% -- Continental Engineering and Aviation Corp.
70% -- Westinghouse Conway R.Co. 7
84% -- Westinghouse PD42-1
68.7% -- General Electric X84A

E. GENERAL

The use of turbojet or ducted fan power plants for direct lift VTOL transport aircraft presents a serious disadvantage for assault transport aircraft. The high velocity, very hot, (over 1000°F) exhaust wake must be positioned or deflected toward the earth for take-off, landing or hovering flight. This condition would:

1. Prohibit the use of this aircraft for take-off and landing operation on most surfaces.

2. Probably require deflecting the exhaust or stopping the engine during an expedited pick-up of personnel or cargo.

3. Probably require the use of fireproof material on the under side of part of the fuselage and/or wing.

4. Present a fire and turbulence hazard to the aircraft and ground personnel.

5. Restrict the use of the aircraft for retrieving personnel or cargo by hoist while in hovering flight.

6. Turbulent exhaust wake may foul and heat the intake air as well as setting gravel in motion for compressor ingestion.

F. ROCKET ON ROTOR SYSTEM

The following information and performance curves on rocket on rotor propulsion systems have been obtained for this study from the Engineering and Research Division of Reaction Motors, Inc., Denville, New Jersey. This information is applicable to both monopropellant and bipropellant systems except where noted. Included among the monopropellants, which were considered are catalytically decomposed monopropellants typified by hydrogen peroxide, and thermally decomposed monopropellants typified by normal propyl nitrate. The bipropellant systems considered applicable are those having simple, dependable ignition characteristics as typified by hydrocarbon fuels with hot, oxidizer rich gases from the decomposition of hydrogen peroxide.

1. Specific Weight Trend

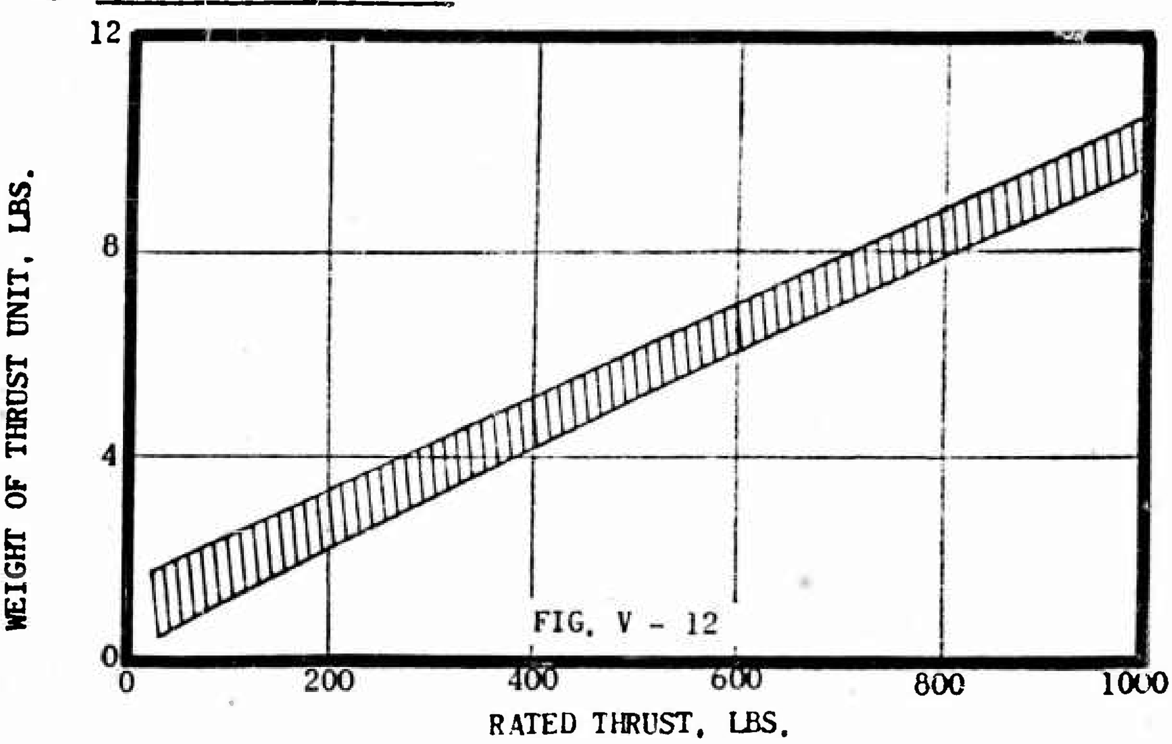


FIG. V - 12

The dry weights of current typical ROR thrust units are shown in Figure V-12 as a function of rated thrust. These are not independent of tip speed since centrifugal loads are a secondary but appreciable source of stress in the ROR thrust unit. In applications to rotors of small diameter and supersonic tip speeds the weight penalty imposed by centrifugal loads may be critical. The weights given in Figure are for approximately 500 g's.

2. Specific Fuel Consumption Trend

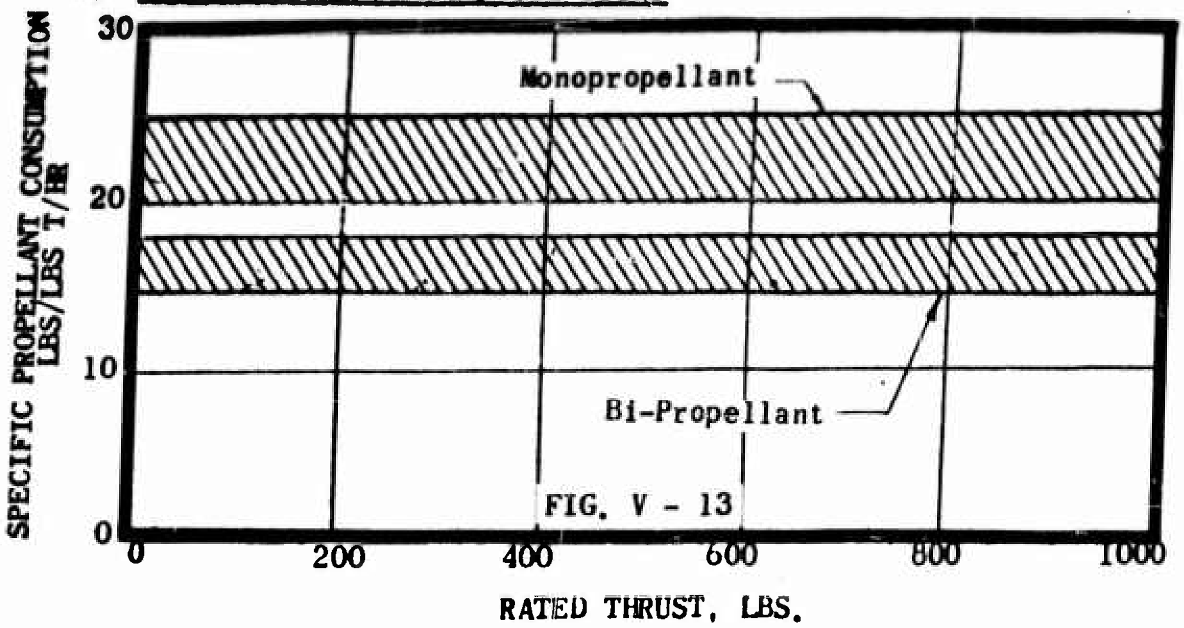


FIG. V - 13

Figure V-13 gives the estimated specific propellant consumption of ROR thrust units. The range of thrust considered applicable to future vertical take-off aircraft extends up to 1,000 lbs. for an individual thrust unit. Separate areas are indicated for monopropellant and bipropellant ROR systems. The lower specific propellant consumption of the bipropellant systems more than compensates for the higher dry weight in ROR systems of high thrust or long duration (above a total impulse of approximately 100,000 lb. secs.).

Specific propellant consumption can be decreased by the use of higher energy monopropellants; however, current higher energy monopropellants are characterized by higher decomposition temperatures which will impose a weight penalty on the thrust units because of lower allowable design stresses.

3. <u>Part Load Fuel Consumption Trend</u>

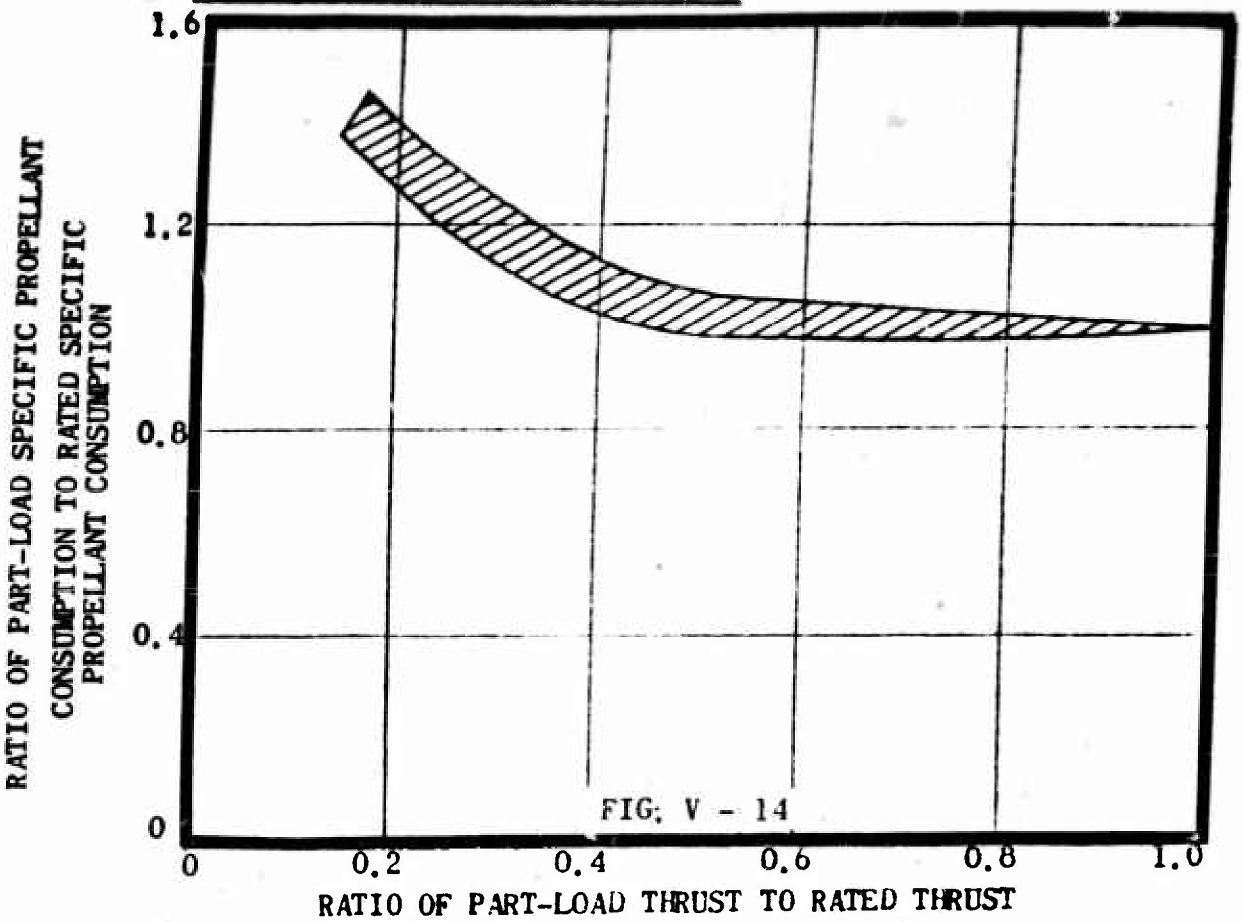


FIG. V - 14

An estimate of reduced thrust requirements on the specific propellant consumption is shown in Figure V-14. A sharp increase in the rate of part load propellant consumption to rated propellant consumption is noted below 40% of rated thrust. This increase in propellant consumption results from nozzle losses due to over-expansion. While this increase in propellant consumption can be avoided by the use of rockets having variable area nozzles, this practice is generally not recommended because of the increased mechanical complexity.

4. Total Dry Weight Trend

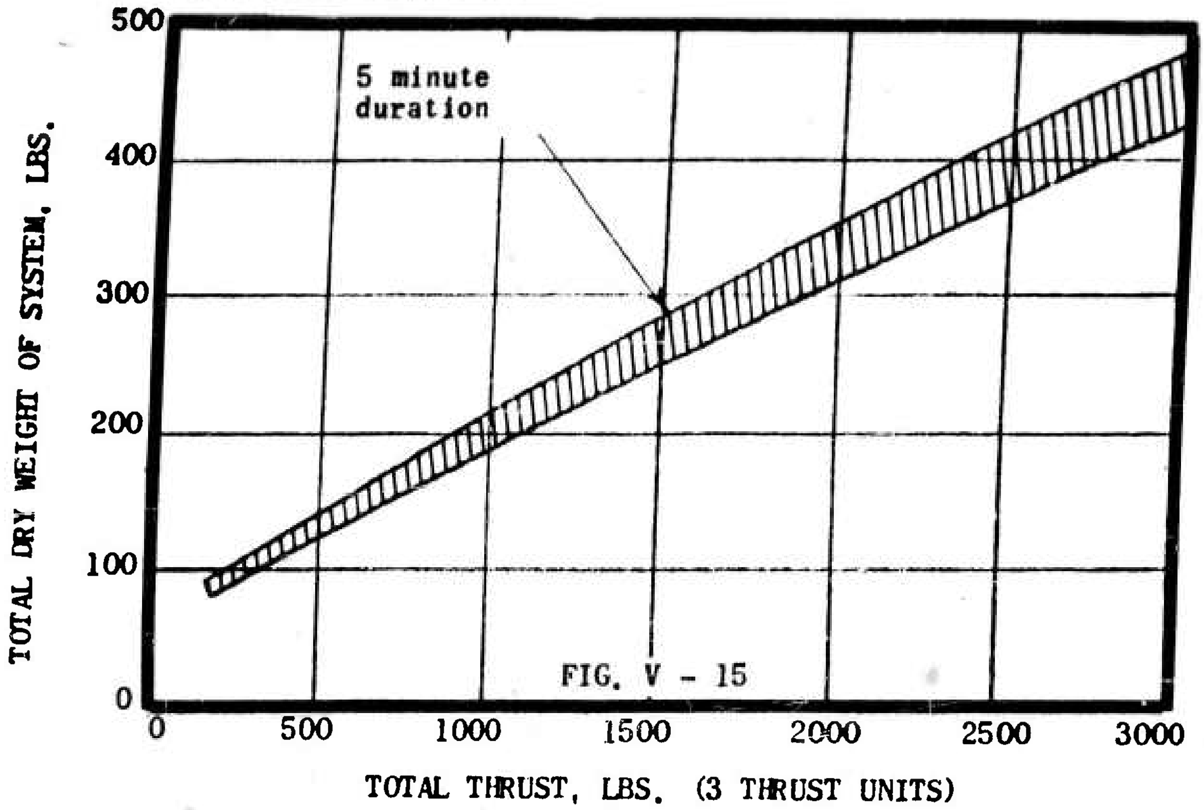


FIG. V - 15

Figure V-15 gives the total dry weight of the system estimated as a function of total rated thrust for maximum duration of five minutes. These weights are based on a three bladed rotor ROR system typified by the system developed by Reaction Motors, Inc., with the cooperation of Sikorsky Aircraft, under BuAer Contract NOa(s)52-1049 for use on the HRS-2 helicopter.

G. PROPELLANT TURBINE SYSTEM

The following information and performance curves for a typical propellant turbine system suitable for driving the rotors of vertical take-off aircraft have been obtained for this study from the Engineering and Research Division of Reaction Motors, Inc., Denville, New Jersey. These curves are generally applicable to the monopropellants and the bipropellant combinations discussed for the ROR systems. Because of the limited operating temperatures permitted by present methods of turbine blade cooling, there is no significant difference shown between the performance ranges of monopropellant and bipropellant turbines.

1. Specific Weight Trend

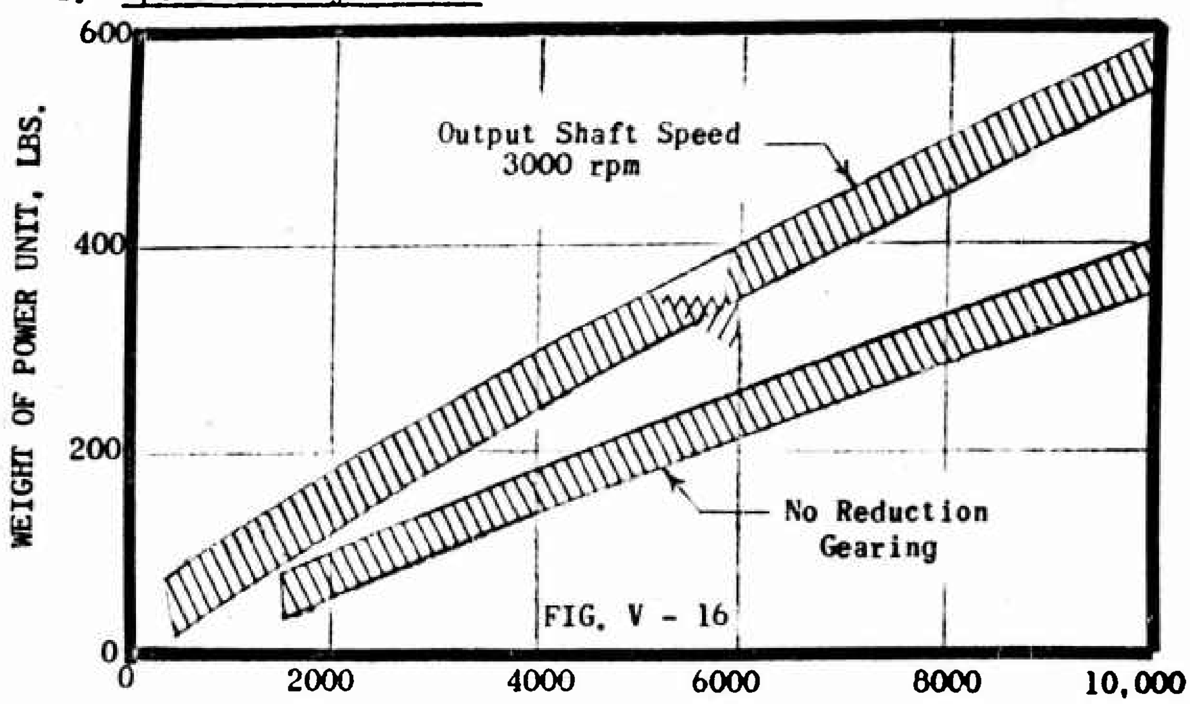


FIG. V - 16

The dry weights of propellant turbine systems are shown in Figure V-16 as a function of rated power. The portions of the propellant turbine systems included with the power unit are the turbine, propellant pump, gas generator and controls, and reduction gears. A separate curve shows the weight of the power unit without the reduction gears. Output shaft speeds without reduction gearing would be approximately 10,000 to 20,000 RPM, the lower speeds occurring in the units of larger rated power.

2. Specific Fuel Consumption Trend

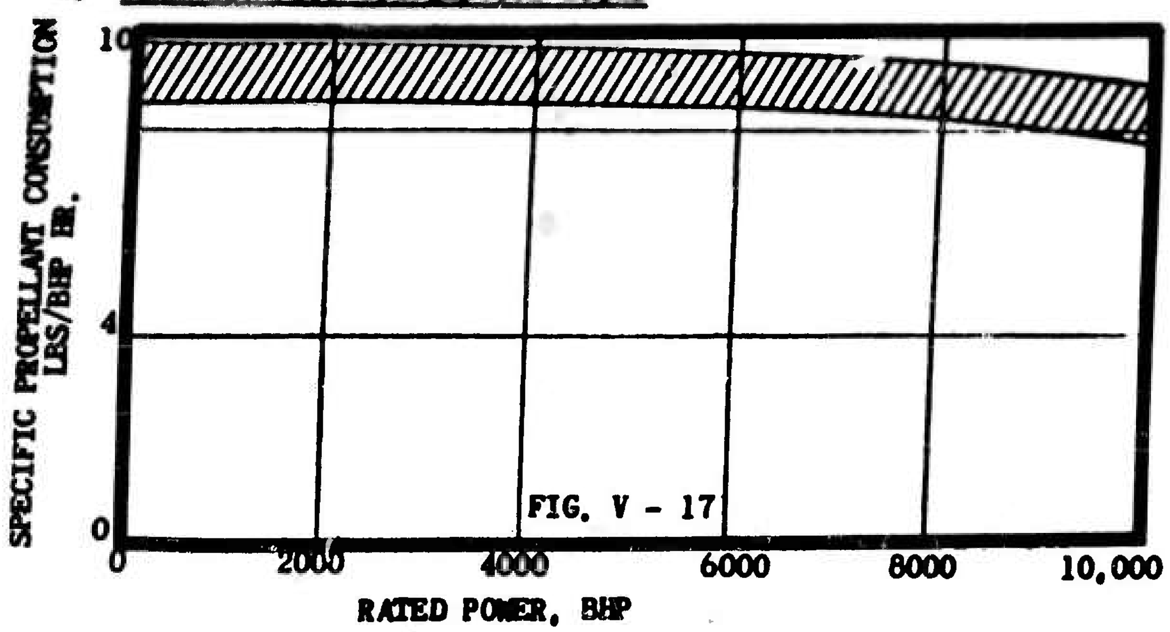


Figure V-17 gives the estimated specific propellant consumption of simple propellant turbine systems as a function of rated power at output shaft speeds of 3,000 to 4,000 RPM.

3. Part Load Fuel Consumption Trend

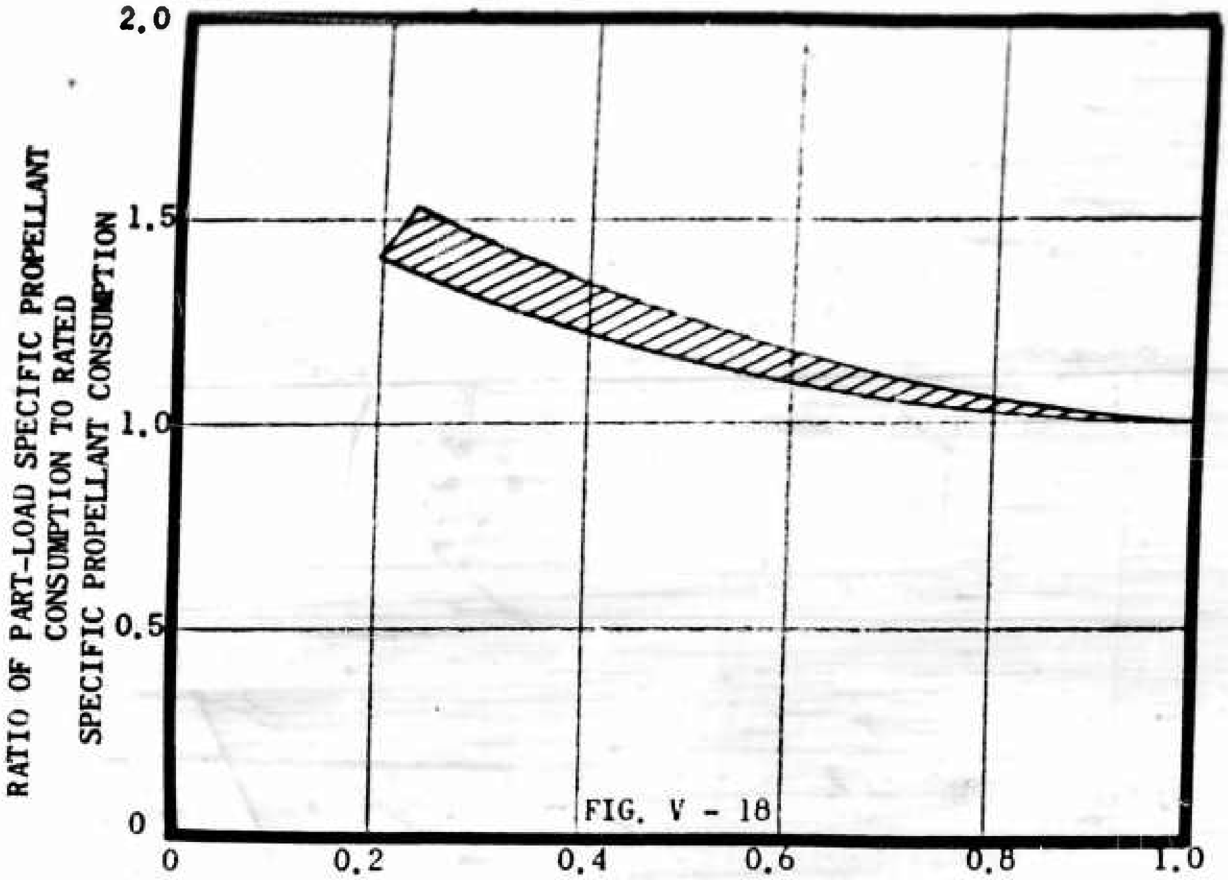


FIG. V - 18

The effect of reduced power requirements on the specific propellant consumption of the simple turbine is estimated in Figure V-18 This curve is similar in form to that shown in Figure V-14 for the ROR system, and many of the same design criteria influence its characteristics. However, unlike the ROR system, sections of turbine nozzle can be blanked off at reduced power to regain rated specific propellant consumption.

4. Total Dry Weight Trend

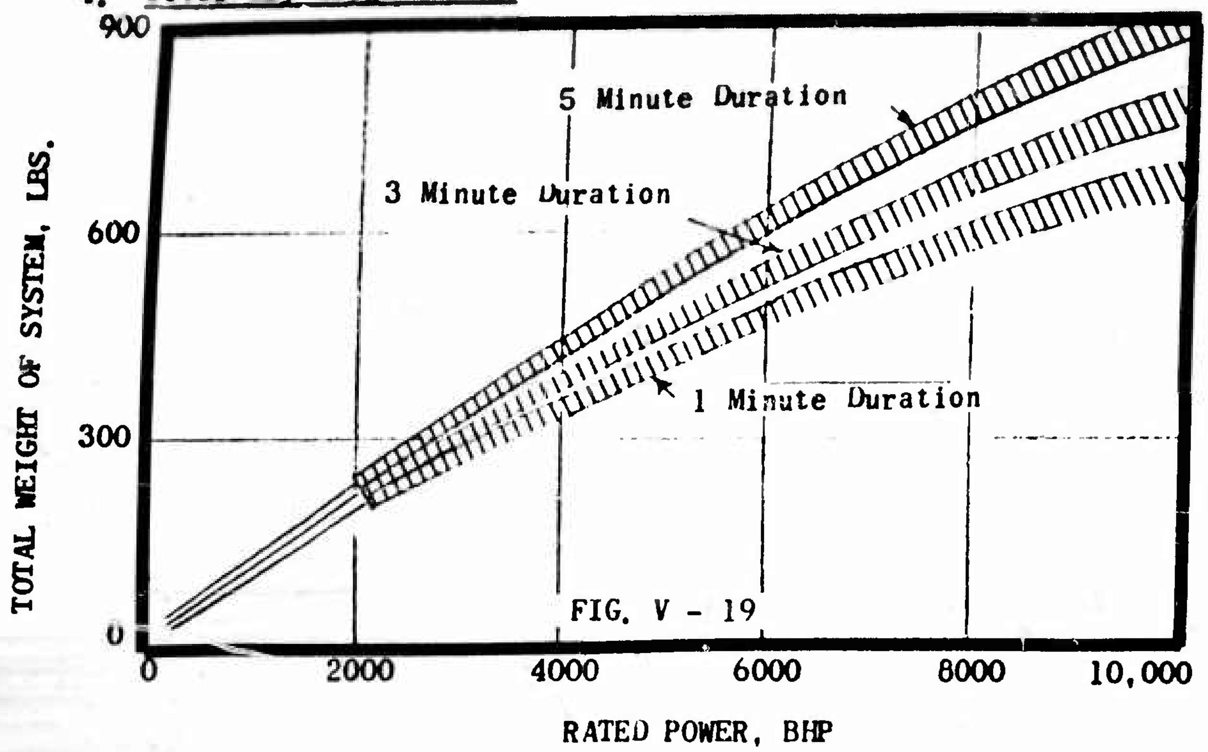


FIG. V - 19

Figure V- 19gives the total dry weight of the propellant turbine system as a function of rated power for maximum run durations of one minute, three minutes, and five minutes. The total dry weight includes the weight of the turbine system itself as represented in Figure V-19 and the weight of propellant tankage, tank supports, lines and valves, etc.

5. Power Variation with Altitude & Ambient Temperature

The power available from a propellant turbine system and the thrust available from a rocket-on-rotor system would increase approximately 2% to 4% with an increase in pressure altitude to 6,000 feet. The influence of the temperature increase to 95°F from standard sea level conditions is negligible.

VI. PERFORMANCE EVALUATION *

A. General Remarks

In order to obtain a direct correlation between the weight trend studies and performance analysis, it was necessary to use the same design parameters in both cases. For this reason, the weight trend expressions of aircraft based on both rotary and fixed wing concepts were modified to include the design parameters used in the performance equations. The hovering performance equations for rotor and propeller type of aircraft were established using the basic equations for this regime of flight as given in Ref. 2 . The same reference was used as a starting point for equations referring to forward flight of rotary wing aircraft. Fixed wing forward flight power required is based on the most part on Ref. 3.

B. Performance Equations - Hovering Flight Regime

1. Ideal Power Loading

Induced power is that power required to compensate for the energy transferred each second to the slipstream when thrust is produced. Assuming the rotor (propeller) is acting as an idealized actuator disc, the minimum power required (induced power) to produce a given thrust would be:

$$HP_{id} = \frac{Tv}{550} = \frac{T}{550}\sqrt{\frac{T}{2\pi R^2 \rho l}}$$

Introducing disc loading, $w = T/l\pi R^2$ into the expression:

$$HP_{id} = \frac{T}{550}\sqrt{\frac{w}{2\rho}}$$

Therefore, ideal power loading may be expressed as:

$$\Lambda_i = \frac{T}{HP_{id}} = \frac{550}{\sqrt{\frac{w}{2\rho}}}$$

and is plotted as a function of disc loading in Fig. VI-1.

2. Actual Power Loading for Non-Ducted Rotors

Due to non-uniform downwash distribution and tip losses, the induced power in hovering is somewhat greater than that expressed for the ideal case. To facilitate the calculation of induced power, the ideal power is increased by a factor K_1, to account for the above losses.

* Prepared by: J. Mallen

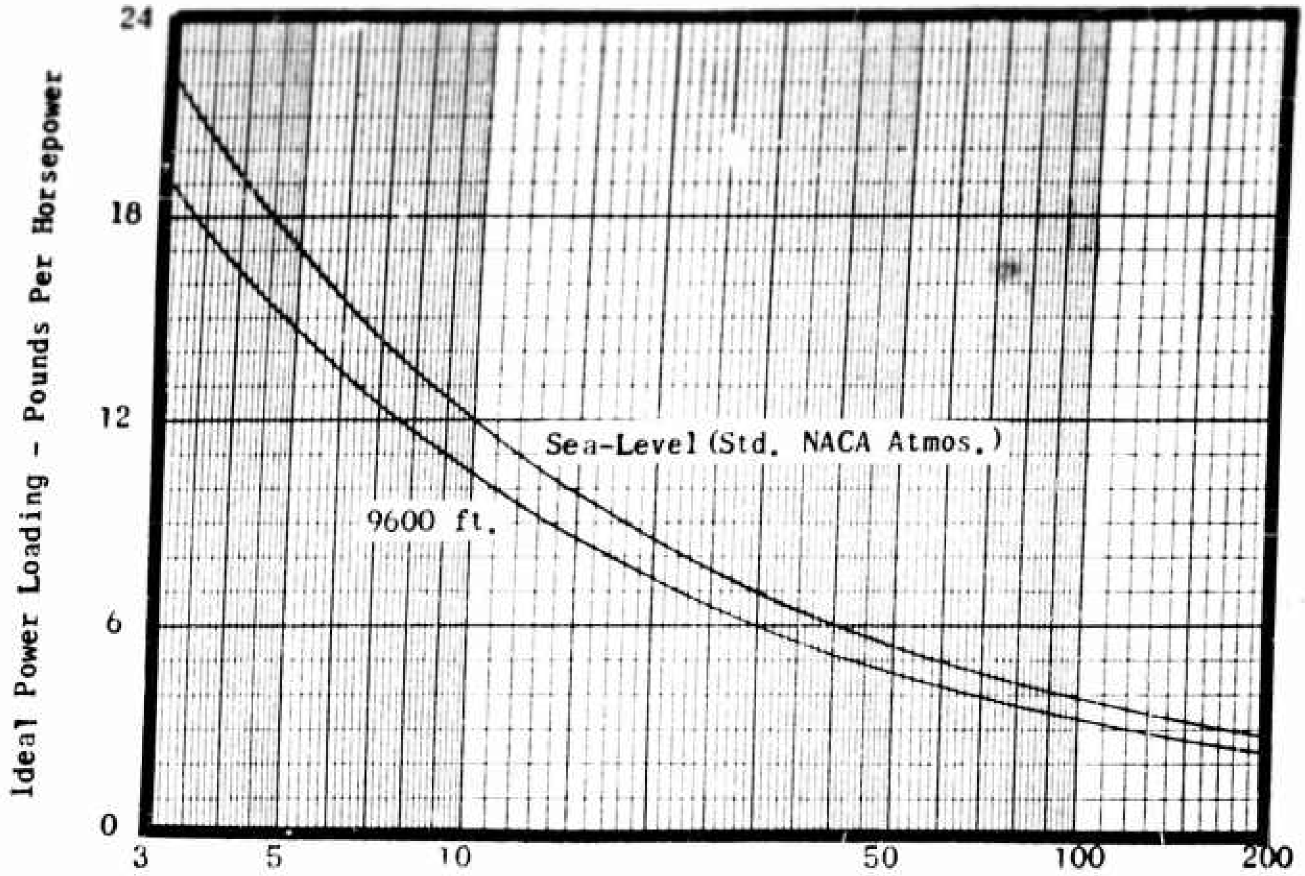


Figure VI-1. Ideal Power Loading In Hovering vs. Disc Loading

In addition, for the case of overlapped rotors, an overlap correction is included in K_1. Therefore, actual induced power may be expressed by:

$$HP_{ind} = \frac{K_1 T}{550} \sqrt{\frac{\omega}{2\rho}}$$

The power required to overcome the profile drag of the blades is defined as:

$$HP_{PRO} = \frac{\sigma i \pi R^2 \rho C_{do} V_t^3}{4400}$$

The variation of profile drag coefficient with average rotor lift coefficient was computed for an NACA 0015 airfoil section. Calculations were made at several tip speeds and at sea level and 9600 feet (pressure altitude of 6000 feet at 95°F ambient temperature). The method of calculation is similar to the procedure used at Vertol Aircraft Corporation for performance estimates which has in the past resulted in good-to-excellent agreement with properly conducted flight tests.

2. Actual Power Loading for Non-Ducted Rotors (Cont'd)

It has been found that conditions at 80% of the blade radius are representative for computing profile power. Consequently, the calculations were based on a profile drag coefficient at .8R. Minimum profile drag was increased by a correction factor of 1.3 to compensate for surface roughness of actual blades. Profile drag coefficients were computed assuming a constant value of Reynold's Number. Mach number effects as well as the effect of changes in density ratio on the drag coefficient were included in the calculation. Profile drag coefficients are plotted in Fig. VI-2A and VI-2b.

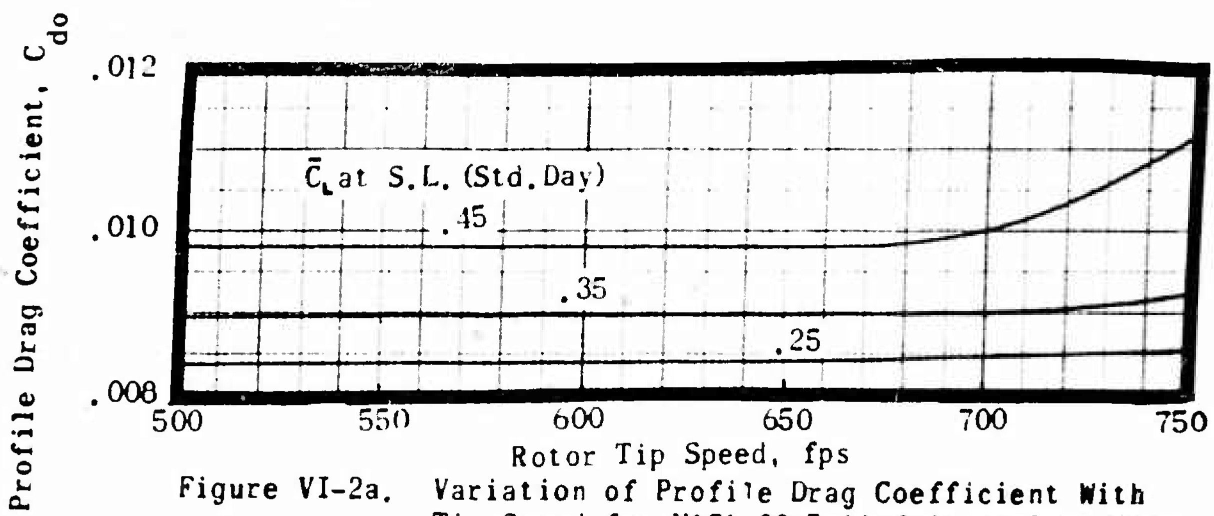


Figure VI-2a. Variation of Profile Drag Coefficient With Tip Speed for NACA 0015 Airfoil at Sea Level and Standard Atmosphere.

Total power in hovering may now be written as:

$$HP_H = \frac{K_1 T}{550}\sqrt{\frac{\omega}{2\rho}} + \frac{\sigma_L \pi R^2 \rho C_{do} V_t^3}{4400}$$

Remembering that average rotor lift coefficient, $\bar{C}_L$ is defined as:

$$\bar{C}_L = \frac{6 \epsilon W}{\sigma_L \pi R^2 \rho V_t^2}$$

and $T = \epsilon W$

where ϵ is the download factor

thus
$$HP_H = \frac{K_1 \epsilon^{3/2} W}{550}\sqrt{\frac{\omega}{2\rho}} + \frac{6 W \epsilon C_{do} V_t}{\bar{C}_L \, 4400}$$

2. Actual Power Loading for Non-Ducted Rotors (Cont'd)

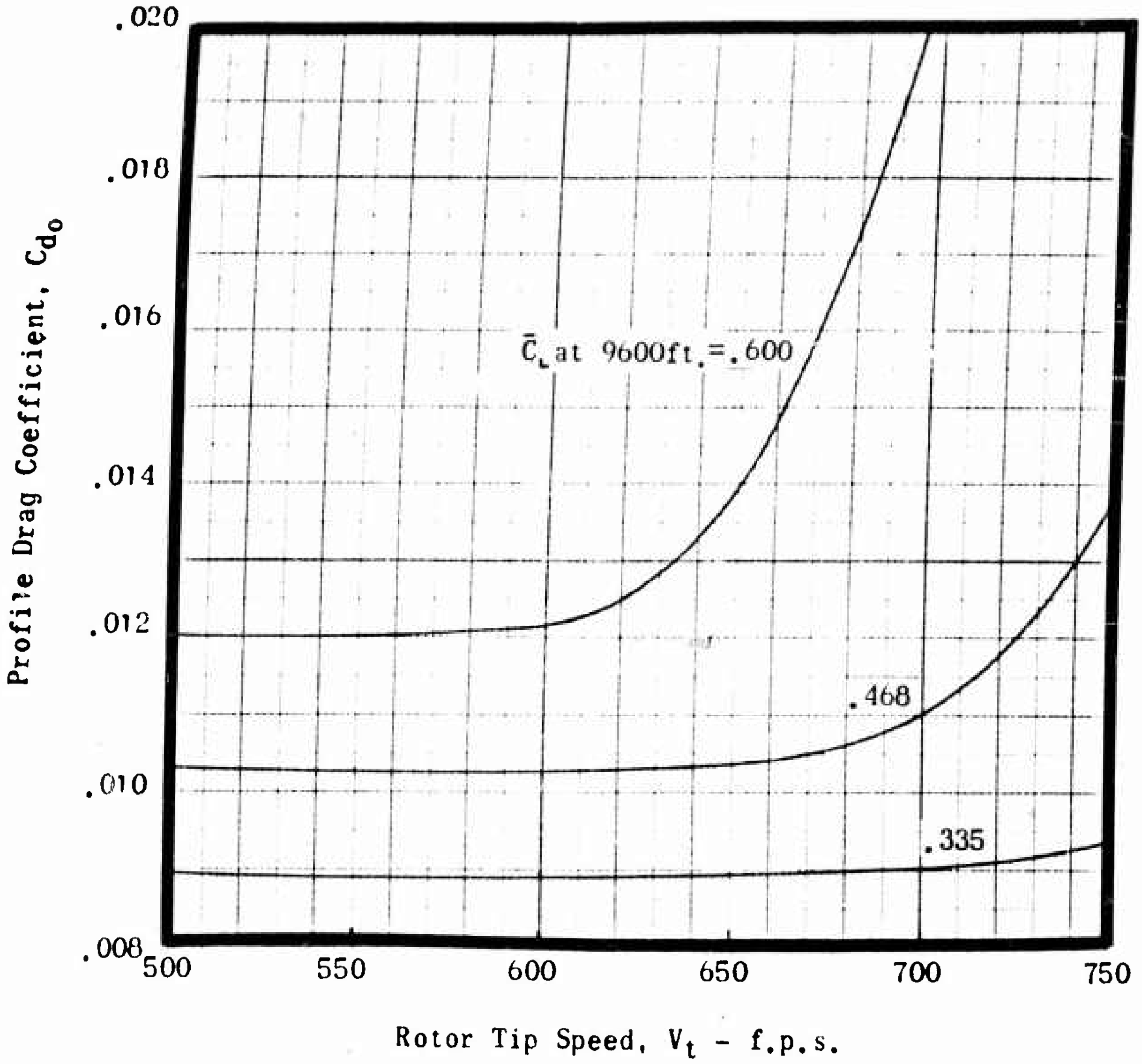


Fig. VI 2-b. Variation of Profile Drag Coefficient with Tip Speed for NACA 0015 Airfoil at an Altitude of 6000 feet and 95°F Ambient Temperature.

Or, actual power loading, Λ_a is:

$$\Lambda_a = \frac{W}{HP_H} = \frac{550}{K_1 \epsilon^{3/2} \sqrt{\frac{\omega}{2\rho}}} + \bar{C}_L \frac{4400}{6 W \epsilon C_{d_o} V_t}$$

2. <u>Actual Power Loading for Non-Ducted Rotors</u> (Cont'd)

For purposes of calculation and presentation, it is convenient to define aerodynamic efficiency as:

$$\eta_{aero} = \frac{\Lambda_a}{\Lambda_\iota}$$

Therefore,

$$\Lambda_a = \Lambda_\iota \, \eta_{aero}$$

The aerodynamic efficiency for non-ducted rotors is plotted in Fig. VI-3a and VI-3b.

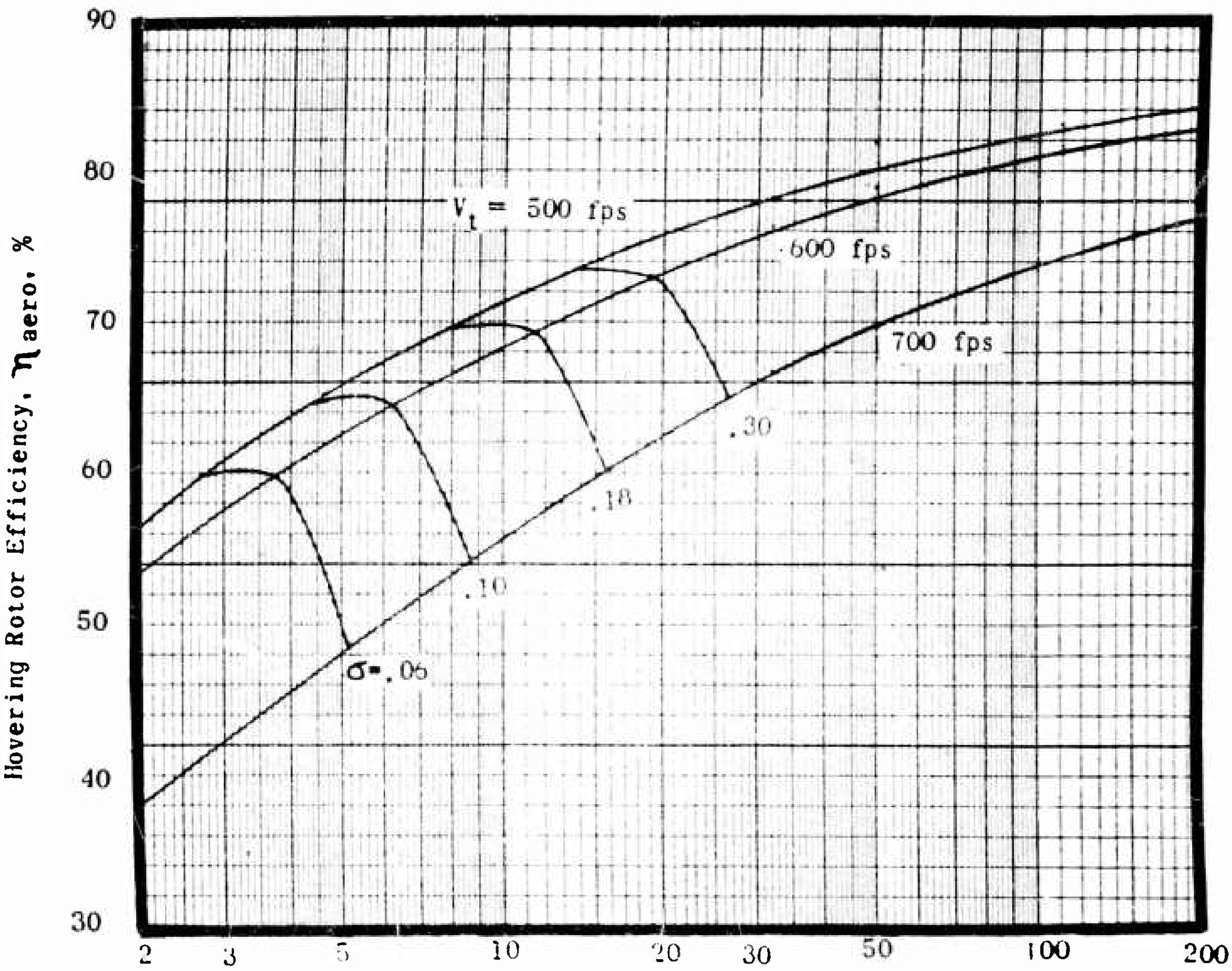


Fig. VI-3a Hovering Rotor Efficiency vs. Disc Loading
NACA 0015 Airfoil (No Overlap) $\bar{C}_L$ = .6
6000' Altitude, Ambient Temp. = 95°F

2. Actual Power Loading for Non-Ducted Rotors (Cont'd)

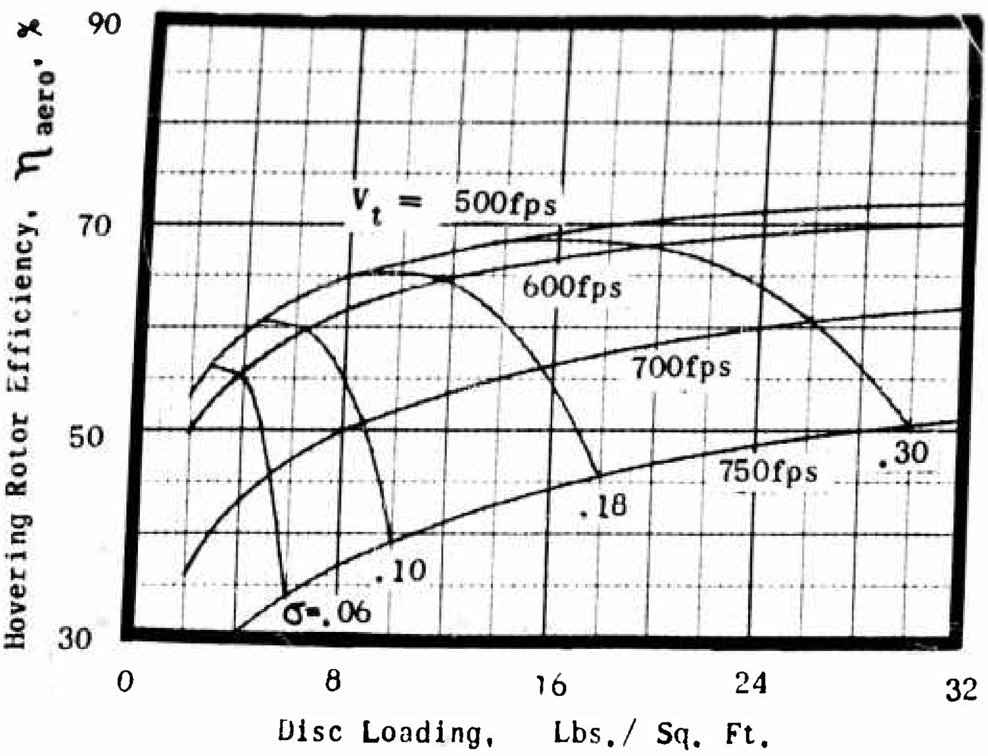


Fig. VI-3b Hovering Rotor Efficiency vs. Disc Loading
NACA 0015 Airfoil (Overlapped Rotors) $\overline{C_L}$ = .6
6000' Altitude, Ambient Temp. = 95°F

3. Actual Power Loading for Ducted Rotors

Actual power loading of ducted rotors was based on NACA test data (Ref. 9). An aerodynamic efficiency factor was determined using the data calculated in Section B-1 with actual test points from Ref. 9. This factor was then applied to the ideal power loading at a density altitude of 9600 ft. The resultant power loading for ducted rotors is presented in Fig. VI-4.

4. Hovering Fuel Requirements

a. General

Hovering fuel requirements were calculated at sea level for a total time of seven (7) minutes. Five minutes were assumed for take-off and landing and two minutes for warm-up. The calculation was based on the hovering power required taking into consideration part-load operation of the powerplants, and includes 10% reserve and 5% increase in specific fuel consumption.

4. Hovering Fuel Requirements (Cont'd)

For convenience, fuel system weight was included with the fuel requirements. Fuel systems are assumed to weigh 0.2 lbs. per gallon for tanks mounted in the fuselage and .15 pounds per gallon for tanks located in the wing. For turboprops, turbojets and by-pass turbojets, JP-4 fuel weighing 6.5 pounds per gallon was used. Therefore, hovering fuel requirements were increased by 1.038 and 1.0231 for tanks in the fuselage and tanks in the wing respectively. For the tip rocket and rocket turbine installations fuel tankage was included in the propulsion group weight.

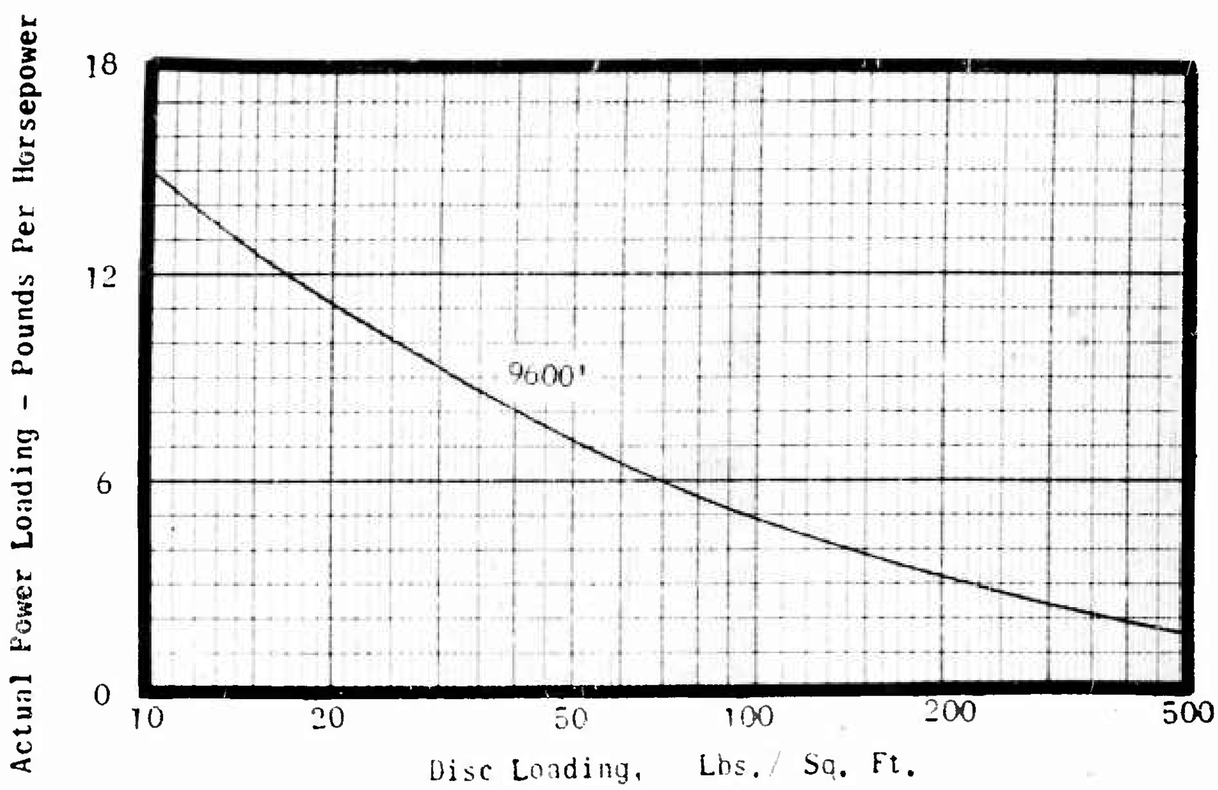


Fig. VI-4 Actual Power Loading vs. Disc Loading
(Shrouded Propellers)

b. Shaft-Driven Turboprop Systems

Hovering fuel requirements for shaft-driven aircraft was calculated as follows:

$$W_{HOV} = \frac{HP_H}{\eta_t} SFC_{MIL}\, \delta \frac{t}{60} \times \frac{1.05}{.9} \times \beta$$

where, HP_M = hovering power.

η_t = transmission efficiency.

t = total time in hovering, minutes.

δ = part-load correction of rated military SFC

β = tankage factor.

Substituting W/Λ_a for HP_H hovering fuel may be expressed in terms of gross weight as:

$$W_{HOV} = .0195 \frac{SFC_{MIL}\, \delta t \beta}{\Lambda_a \eta_t}$$

Since the aircraft is designed to hover at 6000 ft. and 95°F ambient temperature at military power, hovering at sea level under standard atmospheric conditions is accomplished at 67% of rated sea level military power (see Section V, B-4). Consequently, the value of specific fuel consumption is obtained at this part-load power. From Figure V-5, the sea level military static SFC is increased by 1.065.

For the sake of simplicity, the value of Λ_a computed at 6000 ft. altitude and 95°F ambient temperature was used for calculating the hovering fuel requirements at sea level. This assumption results in a conservative (heavy) estimate of fuel required.

c. Turbojet (Direct Lift) Systems

For direct lift applications, wherein the thrust is directed downward, at take-off, hovering fuel requirements may be expressed simply as:

$$W_{HOV} = W\, SFC_{MIL}\, \delta \frac{t}{60} \frac{1.05}{.9} \beta = .0195\, SFC_{MIL}\, \delta t \beta W$$

Hovering at sea level can be accomplished at 72% of the sea level military thrust rating of the turbojet (see Section V, C-4). From Fig. V-8, the value of δ is .90.

For the by-pass turbojet application, the above expression is equally applicable. The part-load fuel consumption correction factor is .955 (see Fig. V-11).

d. Tip Rocket Systems

For the rotary-wing concepts that employ tip rockets at each rotor blade for thrust generation, the hovering fuel requirements at sea level can be expressed as:

$$W_{HOV} = \frac{W}{\Delta_a V_t} 550 \; SFC_{MIL} \frac{t}{60} \times \frac{1.05}{.9}$$

or

$$W_{HOV} = .1605 \; \frac{SFC_{MIL}}{\Delta_a V_t} t W$$

e. Turbine Rocket System

For those concepts that employed turbine rockets to provide shaft horsepower to the rotor, the hovering fuel requirements at sea level were calculated as for the turboprop schemes. However, tankage weight was included in the propulsion group and operation was at a full power. Thus,

$$W_{HOV} = .0195 \; \frac{SFC_{MIL} t}{\Delta_a \eta_t}$$

C. Performance Equations - Forward Flight Regime

1. Conventional Helicopter

Forward flight power required for a conventional rotary wing aircraft is defined as (Ref. 2):

$$HP_{F_{RW}} = \frac{1}{1100 \eta_t} \left[\rho V^3 f + \frac{\sigma \pi R^2 \rho \iota V_t^3 C_{do}}{4} \left(1 + 4\mu^2\right) + \frac{K_2 W^2}{\iota \pi R^2 \rho V} \right]$$

Substituting average rotor lift coefficient and disc loading into the above expression yields:

$$HP_{F_{RW}} = \frac{1}{1100\eta_t}\left[\rho V^3 f + \frac{6 W \epsilon C_{d_o} V_t}{4\bar{C}_L}\left(1+4\mu^2\right) + \frac{K_2 \omega W}{\rho V}\right]$$

Since the fuselage cross-section is defined by the mission payload equivalent flat plate area could be estimated for each concept. Thus, parasite power required could be defined and the remaining portions, profile power and induced power, could be expressed as a function of gross weight. Thus,

$$HP_{F_{RW}} = \frac{\rho V^3 f}{1100\eta_t} + \frac{W}{1100\eta_t}\left[\frac{1.5\epsilon C_{d_o} V_t\left(1+4\mu^2\right)}{\bar{C}_L} + \frac{K_2\omega}{\rho V}\right]$$

2. Compound Helicopter

For the compound helicopter study, wherein the rotor was assumed to be autorotating at an advance ratio of .8 (providing no thrust), power required in forward flight may be expressed as:

$$HP_{F_{CH}} = \frac{\rho V^3 f}{1100\eta_p} + \frac{\rho V^3 S_w C_{d_{ow}}}{1100\eta_p} + \frac{\rho V^3 S_w C_{D_L}}{1100\eta_p} + \frac{\sigma\pi R^2 \rho_\ell V_t^3 C_{d_o}\left(1+5\mu^2\right)}{4400\eta_t}$$

The profile drag correction factor for forward flight has been increased from $4\mu^2$ for the conventional helicopter to $5\mu^2$ for the compound helicopter to account for the higher advance ratio encountered in this design. Substituting the expressions for average rotor lift coefficient and wing area, the above expression reduces to:

$$HP_{F_{CH}} = \frac{\rho V^3 f}{1100\eta_p} + \frac{W}{1100}\left[\frac{2V}{\eta_p\left(C_L/C_{d_o}\right)_w} + \frac{2VC_{L_w}}{\pi \mathcal{R} e\eta_p} + \frac{1.5 V_t C_{d_o}\left(1+5\mu^2\right)}{\bar{C}_L\eta_t}\right]$$

Assuming at $\mu = .8$, $C_{d_o} = .009$ and $\bar{C}_L = .45$ (since $\bar{C}_L$ was substituted for rotor solidity, this value of $\bar{C}_L$ reflects the required solidity in hovering flight), power required in forward flight for the compound helicopter reduces to:

$$HP_{F_{CH}} = \frac{\rho V^3 f}{1100\eta_p} + \frac{W}{1100}\left[\frac{2V}{\eta_p\left(C_L/C_{d_o}\right)_w} + \frac{2VC_{L_W}}{\pi \mathcal{R} e\eta_p} + \frac{.246 V^3}{\eta_t V_t^2}\right]$$

3. Fixed Wing Concepts

Forward flight power required for a fixed wing aircraft is defined as (Ref. 3):

$$HP_{F_{FW}} = \frac{1}{1100\eta_p}\left[\rho V^3 f + \rho V^3 S_W C_{do_W} + \rho V^3 S_W C_{D_i}\right]$$

Substituting $W/q\ C_{LW}$ - for S_W:

$$HP_{F_{FW}} = \frac{\rho V^3 f}{1100\eta_p} + \frac{WV}{550\eta_p}\left[\left(\frac{1}{C_L/C_{do}}\right)_W + \frac{C_{LW}}{\pi \mathcal{R} e}\right]$$

For the tilting ducted propeller concept, the additional power to overcome the drag of the shrouds was estimated. Since the shroud length was assumed to be one diameter, the total area of the shrouds is $i\pi D^2$ where i is the number of shrouds. Substituting for diameter, D in terms of gross weight and disc loading, the total area of the shrouds is W/ω. Assuming a profile drag coefficient of .01, the equivalent flat plate area of the shrouds is .01 W/ω. Thus total power required in forward flight for the tilt wing ducted propeller is:

$$HP_{F_{TDP}} = \frac{\rho V^3 f}{1100\eta_p} + \frac{WV}{550\eta_p}\left[\frac{1}{(C_L/C_{do})_W} + \frac{C_{LW}}{\pi \mathcal{R} e} + \frac{.01\rho V^2}{2\omega}\right]$$

Obviously, for turbojet powered aircraft, assuming drag divergence has not been encountered, the expression for required thrust is:

$$T_j = \frac{\rho V^2 f}{2} + W\left[\frac{1}{(C_L/C_{do})_W} + \frac{C_{LW}}{\pi \mathcal{R} e}\right]$$

4. "Aerodyne" Concept

For the "Aerodyne" concept, it has been assumed that all the lift and propulsive force is derived from the shrouded propellers. After several discussions with Dr. Lippisch of Collins Radio Corporation, the following method of performance evaluation was considered adequate for this study.

The power required in hovering can be expressed as:

$$HP_{H_A} = \frac{W\sqrt{\omega}}{53.7\eta_t\sqrt{\rho/\rho_o}}$$

where, η_t has been assumed to be .88 for this design concept. Considering the losses associated with deflecting the thrust for the hovering regime of flight, the above expression yields values for hovering power which are in good agreement with the NACA test results for shrouded propellers (Ref. 9).

In forward flight, the propellers supply the lift (equal to gross weight) and the thrust required to overcome drag. Thus, the resultant thrust of the shrouded propeller, T_R may be expressed as:

$$T_R = \sqrt{W^2 + D_f^{\,2}}$$

where, D_f is the total drag of the aircraft in forward flight.

Assuming the length of the shroud to be 2D, the total shroud area is equal to $i\pi D \times 2D = 2i\pi D^2$ where i is the number of shrouds. Estimating an equivalent flat plate area for the remaining portions of the aircraft f, total drag may be expressed as:

$$D_f = q f + 2 i \pi D^2 q\, C_f$$

where, C_f is equal to skin friction drag coefficient = .005.

Thus, the resultant thrust required, T_R may be calculated for each forward flight condition.

Power required in forward flight, based on data obtained from Dr. Lippisch, may now be expressed as:

$$HP_{F_A} = \frac{T_R\sqrt{\omega_f}}{80\eta_t\sqrt{\rho/\rho_o}}$$

where, ω_f is the disc loading in forward flight $= 4T_R/i\pi D^2$ and η_t is equal to .94.

5. Forward Speed Limitations for Rotary-Wing Aircraft

Level flight forward speed of the helicopter can be limited by power as well as either stalling of the retreating blade or compressibility effects on the advancing blade.

5. Forward Speed Limitations for Rotary-Wing Aircraft (Cont'd)

a. Compressibility

Compressibility effects on the advancing blade have not been encountered in flight tests conducted by Vertol Aircraft Corporation (in autorotative flight) up to an estimated 1.25 Mach critical at the tip of the advancing blade. Compressibility limitations to forward speed were conservatively based on this experience and are indicated in Fig. VI-5.

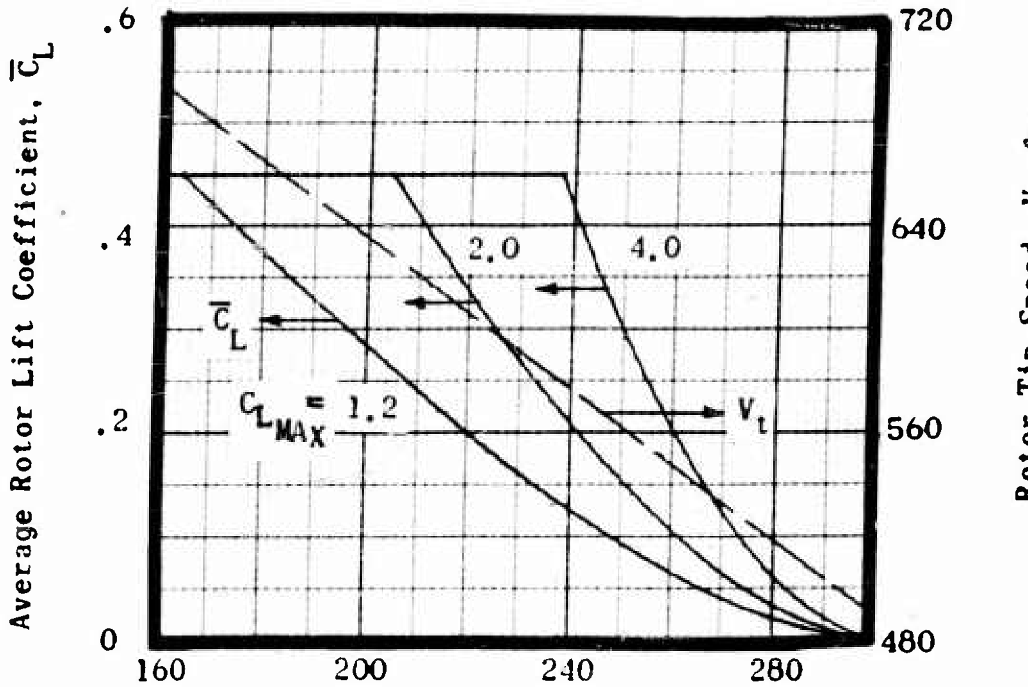


Fig. VI-5 Retreating Blade Stall and Compressibility Speed Limitations @ Sea Level, Std. NACA Day.

b. Retreating Blade Stall

The method employed in determining the speed at which retreating blade stall occurs is based on actual flight test data for tandem and single rotor helicopters.

As the forward speed of a helicopter increases, the retreating blade of the helicopter must operate at higher lift coefficients in order to compensate for the lower resultant air velocity. By analogy with fixed-wing, the resultant retreating blade stalling velocity can be expressed as:

$$V_{R_{MIN.}} = \sqrt{\frac{2 K_s w_B}{C_{L_{MAX}} \cdot \rho}}$$

where, w_B should actually be the blade loading in the stalled portion. Since it is impractical to calculate the actual value of w_B at stall an empirical factor, K_S, has been used to correlate the formula with test results. For current helicopter models, a value of $K_S = 3$ provides good correlation with flight test results based on a $C_{L_{max}} = 1.2$.

Thus, $V_{R_{MIN.}} = \sqrt{\frac{6 w_B}{C_{L_{MAX}} \cdot \rho}}$ and, $V_S = V_t - V_{R_{MIN}}$

For maximum speed in cruising the stall speed has been reduced by 10%. Retreating blade stall limitations are shown in Fig. VI-4 for a C_{Lmax} of 1.2.

Based on a boundary layer control program presently under development by this contractor, the forward speed capabilities of the helicopter equipped with BLC rotors has been estimated. The boundary layer control proposed is of the circulation control or blowing type. Using the empirical relationship developed above, C_{Lmax} required for a given speed has been calculated assuming an average rotor lift coefficient in hovering at sea level of .45. Rotor tip speed was reduced as forward speed increased commensurate with the compressibility limit criterion. These results are presented in Fig. VI-5. It should be noted that even higher forward speeds could be attained by decreasing the average rotor lift coefficient with consequent increase in rotor weight and decrease in hovering efficiency. For the purposes of this study, however, the assumed value of $\overline{C_L}$ provided sufficient information to indicate the possible extension of the helicopter speed potential.

Power required for BLC was estimated as a function of maximum lift coefficient, C_{Lmax}, and expressed as a percentage increase in basic cruise power. The variation with C_{Lmax} was assumed linear between a C_{Lmax} of 1.2 where no additional power for BLC was required to a 10% increase in power at a C_{Lmax} of 4.0. The additional power was assumed to be supplied by the helicopter main engines eliminating any need for auxiliary power units.

c. Power Limits

Since these aircraft are designed to meet high hovering performance, which results in low installed power loading, forward flight speed will not be limited by available power.

6. Range Fuel Requirements

a. General

Range fuel requirements were calculated for a total distance of 850 statute miles cruising at sea level. The calculation was based on the cruise power required at take-off gross weight; i.e., no variation in gross weight was considered as fuel is consumed. Both of these assumptions result in conservative (heavy) estimates of required fuel. It was further assumed that the required cruise power is obtained from the engine(s) operating at 80% of its (their) rated military power. This latter assumption obviously requires shutting down a sufficient number of power sections. A 10% fuel reserve and a 5% increase in specific fuel consumption was included. As for the hovering fuel required calculation, the fuel system was included in the estimation for required cruising fuel.

Therefore, range fuel requirements can be expressed as follows:

$$W_{CR} = HP_F \; SFC_{MIL} \; \delta \frac{L}{V} \times \frac{1.05}{.9} \beta$$

where,

L is the total distance traveled in feet

V is the cruise speed in fps

δ is the part-load (80%) correction of rated military SFC taking into consideration cruising speed

HP_F is the cruise shaft horsepower required

Substituting for L and reducing,

$$W_{CR} = 1455 \frac{HP_F \; SFC_{MIL} \; \delta \; \beta}{V}$$

Since the fuselage cross-section was determined by the mission payload requirements, the equivalent flat plate area was estimated for each concept from preliminary layouts. Therefore, the fuel required to overcome parasitic drag could be assessed directly at each speed. The remaining portions of fuel required to overcome rotor, wing, or shroud profile drag and rotor, or wing induced drag were obtained as a function of gross weight.

To determine the optimum combination of aerodynamic parameters for establishing the take-off gross weight as a function of cruise speed, the following items were varied for each design concept:

DESIGN CONCEPT	ITEMS VARIED
a. Rotary Wing Concepts	
1. Conventional Tandem Rotor Helicopter	ω, V_t, $\bar{C}_L$
2. Tandem Rotor Helicopter equipped with BLC Rotors	ω, V_t, $C_{L_{MAX}}$
3. Compound Helicopter	ω, V_t, C_{L_ω}, AR
4. Retractoplane	ω, V_t, C_{L_ω}, AR
b. Fixed-Wing Concepts	
1. Tilt Wing Propeller	ω, V_t, C_{L_ω}.
2. Tilt Wing Turbojet	C_{L_ω}, AR
3. Tilt Wing By-Pass Turbojet	C_{L_ω}, AR
4. Tilting Ducted Propeller	ω, C_{L_ω}, AR
5. Special Hovering Turbojet	C_{L_ω}, AR

DESIGN CONCEPT	ITEMS VARIED
b. Fixed-Wing Concepts (Continued)	
6. Deflected Turbojet Thrust	C_{L_W}, AR
7. Deflected By-Pass Turbojet Thrust	C_{L_W}, AR
8. Vectored Lift	w, V_t
9. Breguet-Kappus	w, AR
10. Aerodyne	w

For the rotary wing concepts, disc loading, w was varied from 4 to 12 lbs/ft.^{2} and rotor tip speed, V_t was varied from 550 to 700 fps. A maximum value of $\bar{C}_L$ of .45 at sea level was used to assure adequate control for all flight regimes. This value was reduced where necessary to prevent retreating blade stall for the conventional tandem rotor helicopter. Aspect ratio, AR, was varied from 6 to 10 and operational wing lift coefficient, C_{LW} was varied from .4 to .8 for both the compound helicopter and retractoplane designs.

For fixed wing designs, aspect ratio was varied from 4 to 10 and operational wing lift coefficient from .4 to .8. For the tilt-wing propeller and vectored lift designs, disc loading was varied from 40 to 100 lbs/ft.^{2}. The disc loading for the Breguet-Kappus and Aerodyne aircraft was varied from 70 to 200 lbs/ft.^{2}. Propeller tip speed was varied from 600 to 800 fps. for all applicable designs.

The span of the tilt-wing propeller and vectored lift designs was determined by the disc loading and number of propellers since it was assumed the entire wing would be immersed in the propeller slipstream. Derivation of the span as a function of these parameters appears in a following section.

For the vectored lift concept, full span Fowler flaps were assumed. The chord to propeller diameter ratio was established as .5 based on NACA tests (Ref. 4). Assuming the extended wing chord to be 30% greater than the wing chord with flaps retracted and using the above ratio for chord to propeller diameter, the wing area was a function of disc loading. Consequently, for this design concept, the operating wing lift coefficient was not an independent variable.

Several basic aerodynamic factors were assumed to be constant throughout the study. Equivalent flat plate area was estimated to be 67 sq. ft. for the tandem rotor helicopters, 56 sq. ft. for the compound helicopters and 36 sq. ft. for the retractoplane concepts. The remaining fixed wing concepts were estimated to have an equivalent flat plate area of 34 ft.^{2}.

For those concepts that are shaft driven, a 6% gear loss was used in estimating required engine power. An additional 7% loss for anti-torque power was assumed in hovering for all applicable designs. A propeller efficiency in forward flight of 85% was used throughout the study.

An induced power factor in hovering, K_1, of 1.12 was used for all isolated rotors. A value of 1.23 was used for the tandem rotor helicopter. The induced power in forward flight, K_2 was 1.12 for single rotor designs and 1.8 for tandem rotor designs. A wing efficiency factor, e, of 85% was used for all fixed wing concepts. Values of wing profile drag was based on the NACA 2415 airfoil section.

C. Weight Trends for Performance Evaluation

1. General

In Section IV and V, expressions for determining aircraft component weights were developed from statistical data using pertinent design parameters for correlating factors. Many of these design parameters used in the weight trend correlations can be related to each other. Therefore, to minimize the number of variables, the weight trend expressions were manipulated so that only a few of the basic aerodynamic parameters remained. Furthermore, to simplify the calculation of take-off gross weight, it was desirable to express all weight items either as a constant number of pounds or as a direct function of gross weight. Since several correlating factors developed in Section IV involve gross weight to a fractional power it was necessary to linearize these terms. This was done simply by solving for a new constant at an assumed gross weight, W.

The derivation of the weight trend expressions used in evaluating the take-off gross weight is described in the following expressions.

2. Rotary Wing Concepts

a. Rotor Group

From Section IV-B.1, the rotor hub and hinge weight is expressed as:

$$W_{HH} = 92.4\, K^{.5287}$$

where, $K = \dfrac{W R\, HP}{10^{7}}$

Rotor radius, R may be expressed as a function of gross weight, W; disc loading, w and number of rotors, i. Thus,

$$R = \sqrt{\frac{W}{i \pi w}}$$

Horsepower may also be expressed as a function of gross weight and actual power loading, Λ_a

$$HP = \frac{W}{\Lambda_a}$$

Substituting, for rotor radius and horsepower, rotor hub and hinge weight is:

$$W_{HH} = 92.4 \left[\frac{W^{5/2}}{(i \pi w)^{1/2} \Lambda_a 10^{7}} \right]^{.5287}$$

Since W is raised to a fractional power, a value of gross weight, $\overline{W}$ was selected and the value of k_W calculated so that: $k_W = \dfrac{\overline{W}^{1.3217}}{\overline{W}}$

Thus

$$W_{RH} = 92.4 \left[\frac{1}{(i\pi w)^{1/2} \Lambda_a 10^7}\right]^{.5287} k_w W$$

The rotor blade weight correlation can also be written in terms of basic aerodynamic parameters. Starting from the expression developed in Section IV-B.2:

$$W_B = 226\ K^{.6299}$$

where,
$$K = \frac{W R^2 \sigma}{V_t \times 10^2}$$

Rotor solidity, σ may be expressed in terms of disc loading, rotor tip speed, V_t and average rotor lift coefficient $\bar{C}_L$. Thus,

$$\sigma = \frac{6 \epsilon w}{\rho V_t^2 \bar{C}_L}$$

where, ϵ = download factor

Substituting,

$$W_B = 226 \left[\frac{6\epsilon W^2}{i\pi\rho V_t^3 \bar{C}_L 10^2}\right]^{.6299}$$

or

$$W_B = 226 \left[\frac{6\epsilon}{i\pi\rho V_t^3 \bar{C}_L 10^2}\right]^{.6299} k_w W$$

where
$$k_W = \frac{W^{1.2598}}{\bar{W}}$$

b. <u>Rotor Drive System</u>

Drive system per rotor was expressed as:

$$W_D = 320\ K^{.6741}$$

where

$$K = \frac{HP}{\Omega}$$

and Ω = Rotor RPM

Substituting for rotor rpm, $\Omega = 60V_t/2\pi R$, the drive system weight is:

$$W_D = 320\left[\frac{W^{3/2}\,2\pi}{\Lambda_a\,60\,V_t\sqrt{\iota\pi w}}\right]^{.6741}$$

or

$$W_D = 320\left[\frac{.1047}{\Lambda_a\,V_t\sqrt{\iota\pi w}}\right]^{.6741} k_w\,W$$

and

$$k_W = \frac{W^{1.011}}{\overline{W}}$$

c. Shafting

For the gas turbine powered retractoplane and compound helicopter designs, the powerplants were assumed to be mounted in wing nacelles; requiring a synchronizing shaft to transmit the power to the rotor drive system. The weight of this item as developed in Section IV-H.3 is:

$$W_{PS} = 63\left(\frac{HP}{\Omega_P}\right)^{.5} L$$

where Ω = shaft rpm (assumed to be 6000 rpm)
L = length of shafting, ft.
HP = horsepower per shaft = 1/2 total horsepower

The length per shaft was assumed to be 1/4 of the wing span. Thus, total length is:

$$L = \frac{2}{4}b_W = \frac{1}{2}\sqrt{ARS} = \frac{1}{2}\sqrt{AR\frac{W}{\gamma C_L}}$$

Thus,

$$W_{PS} = \frac{6.3}{2}\left[\frac{W}{\Lambda_a\,2\times6000}\right]^{.5}\sqrt{\frac{ARW}{\gamma C_L}}$$

or,

$$W_{PS} = 0287\left[\frac{AR}{\gamma C_L\Lambda_a}\right]^{.5} W$$

d. Propeller Synchronizing Transmission

The gearing necessary to connect the shafting to the propeller is:

$$W_{SX} = 130\left(\frac{HP}{\Omega}\right)^{.5} N$$

where, N is the number of propeller nacelles = 4. Thus,

$$W_{SX} = 130\left(\frac{W}{4\Lambda_a\,6000}\right)^{.5} 4$$

where, $k_W = \dfrac{W^{.5}}{\overline{W}}$

e. Powerplants

Powerplant weight was correlated on the basis of military static power, or thrust delivered at sea level under standard atmospheric conditions. Thus, for shaft power units wherein the required power was dictated by hovering at 6000 ft. at 95°F ambient temperature the weight of the powerplant is

$$W_{eng.} = \omega_{eng.} \, HP$$

where, $\omega_{eng.}$ is the specific weight in pounds per horsepower.

Thus,

$$W_{eng} = \omega_{eng.} \frac{W \delta}{\Lambda_a \eta_t}$$

where, δ = is the power correction factor to account for losses at 6000' and 95°F.

For those designs, such as the tip rocket or turbine rocket powered rotor compound helicopter and retractoplane designs, the turboprop engine was designed for cruise at 80% of military static sea level operation. Thus,

$$W_{eng.} = \frac{\omega_{eng.} \, HP_F}{.80}$$

It may be recalled that cruise power, HP_F was derived as a function of a constant value of equivalent flat plate area and the remaining portion was a function of gross weight. Consequently, for this case, powerplant weight was expressed as a given number of pounds (that portion to overcome parasitic drag) in addition to a percentage of gross weight (to overcome rotor and wing profile and induced drags).

The tip rocket rotor engine weight was calculated as:

$$W_{eng.} = \omega_{eng.} \frac{550 \, \delta}{\Lambda_a V_t}$$

f. Propellers

For the compound helicopter and retractoplane propeller driven aircraft, conventional propellers were used. The weight was correlated as .20 lbs. per propeller horsepower. Consequently, propeller weight was expressed as a function of cruise power multiplied by 1.25 since the cruise condition was assumed to be at 80% of military power.

g. Body Group

Body group weight trend was correlated as a function of gross weight squared times the fuselage surface area; S_F. The surface area for each rotary-wing concept was approximated in terms of rotor radius and fuselage circumference as determined by the cargo space required. From preliminary layouts, a circumference of 39 ft. was necessary for both the tandem rotor helicopter and compound helicopter designs. The retractoplane, due to the space required for retracting the rotor, required 42 ft.

For the tandem overlapped rotor design, the length of the fuselage including nose enclosure and aft pylon was determined to be 1.71R. Thus, S_F can be expressed as:

$$S_F = 1.71\, R \times 39 = \;\; .4 \sqrt{\frac{W}{i \pi w}}$$

Body weight for the tandem helicopter is then:

$$W_F = 1.7 \times 496 \left[\frac{W^2\, 63.4}{10^{10}} \sqrt{\frac{W}{i \pi w}} \right]^{.341}$$

or

$$W_F = 1.352 \left[\frac{1}{\sqrt{i \pi w}} \right]^{.341} k_W\, W$$

For the compound helicopter designs, fuselage length was approximated by 1.64R. Surface area was then 1.64R (.8 x 39) where the .8 factor accounts for the decreased required circumference of the tail boom. Thus, body group weight for the shaft-driven rotor compound helicopters is:

$$W_F = 1.26 \times 496.3 \left[\frac{W^2 \times 51.15}{10^{10}} \sqrt{\frac{W}{i \pi w}} \right]^{.3407}$$

or

$$W_F = .935 \left[\frac{1}{i \pi w} \right]^{.3407} k_W\, W$$

For the tip rocket rotor compound helicopter, W_F is

$$W_F = .854 \left[\frac{1}{i \pi w}\right]^{.3407} k_w \; W$$

Fuselage length of the retractoplane designs was estimated to be 1.68R. Thus S_F is equal to 1.68R (.8 x 42) and body weight for the shaft driven versions is:

$$W_F = .934 \left[\frac{1}{i \pi w}\right]^{.3407} k_w \; W$$

and for the tip rocket rotor retractoplane,

$$W_F = .849 \left[\frac{1}{i \pi w}\right]^{.3407} k_w \; W$$

where $k_w = W^{.852} / W$ for all the above expressions.

h. Wing Group

The expression for basic wing weight from Section IV-B1 is:

$$W_{BW} = 41.57 C_1 (S)(W_W)^{1/4} + \frac{.6 (LF)\, b^3 (W_W)\, TF}{f}$$

where,
- S = wing area, ft^2
- W_W = wing loading, lbs/ft^2
- b = wing span, ft^1
- C_1 = non-bending material factor = .024
- L.F = load factor = 4.5
- T.F = taper factor
- f = 35000 lbs/in^2

Substituting for wing area $W/q\,C_L$ and for span $\sqrt{ARS}$ the expression for basic wing weight is

$$W_{BW} = \frac{W}{(q C_L)^{.75}} + .000772 \; T.F \left(\frac{AR\, W}{q C_L}\right)^{3/2} q C_L$$

or

$$W_{BW} = \left[\frac{1}{(q C_L)^{.75}} + .000772 \; T.F \frac{(AR)^{3/2}}{(q C_L)^{1/2}} \; k_w\right] W$$

where $k_w = \frac{W^{3/2}}{W}$

For all rotary wing concepts that require a fixed wing, the taper factor was calculated from,

$$TF = \left(\frac{1+\lambda}{6}\right)\left(\frac{.51 K_R + \lambda^2 K_T}{K_R^2 + K_R K_T \lambda}\right)$$

where, the following values were assumed:

K_R = thickness of root chord expressed in percent = .16
K_T = thickness of tip chord expressed in percent = .08
λ = equivalent tip chord/root chord = .5

and T.F. = .794

i. Miscellaneous

The remaining items such as alighting gear, tail group, flight controls, etc. comprising either weight empty or fixed useful load were correlated as a direct function of gross weight or a fixed number of pounds. Consequently, these items did not require any modification.

3. Fixed Wing Concepts

a. General

Derivation of the weight expressions for the fixed wing concepts were obtained using the same procedure and definitions of basic aerodynamic parameters as for the rotary wing aircraft. Consequently, it is not deemed necessary to repeat these derivations. Instead the salient features and assumptions of each design concept is briefly discussed.

A summary of fixed and rotary wing weights terms used for performance evaluation appears at the end of this section.

b. Fixed Wing Concepts

Rotor group trends were obtained as for the rotary wing concepts. For the tilting ducted propeller aircraft the surface area of the shroud was obtained by assuming the shroud length to be equal to one diameter and the circumference to be πD. The shroud length for the "Aerodyne" was taken as 2D, thus the surface area is equal to $2i\pi D^2$

c. Drive System

Under drive system, several of the concepts required propeller extension shafts. The correlating factor, K for this item is the same as previously derived for the propeller synchronizing transmission. The extension shaft length was assumed to be .5D for the tilt-wing propeller and vectored lift designs. For the "Aerodyne" and tilting ducted propeller, the length was taken to be 1/4 of the propeller diameter. The length for the Brequet-Kappus VTOL concept was assumed to be equal to one foot.

Propeller synchronizing shafting was deemed necessary in case of powerplant failure for the tilt-wing propeller, vectored lift, tilting ducted propeller, "Aerodyne" and Breguet-Kappus concepts.

Since the wing is assumed to be entirely immersed in the propeller slipstream for the tilt-wing propeller and vectored lift aircraft, the length of the synchronizing shaft was expressed in terms of propeller disc loading and number of engines. Thus, wing span, based on slipstream effects of Ref. 4, is:

$$b = b_p\, d_1 + \text{fuselage width} + \text{dist. of inboard propellers to fuselage}$$

where, b_p = number of propellers

d_1 = slipstream diameter = $\sqrt{D^2\left(\frac{1}{1+K_p}\right)}$

and K_p = .707 when the propeller extension length to propeller diameter is .5.

Assuming the width of the fuselage plus the distance of the inboard propellers to the fuselage can be approximated by D, the wing span is

$$b = 2 b_p \sqrt{\frac{W}{b_p \pi w}} \sqrt{\frac{1}{1+K_p}} + 2\sqrt{\frac{W}{b_p \pi w}}$$

or

$$b = \sqrt{\frac{W}{b_p w}} \left[.865\, b_p + 1.13 \right]$$

The length of the synchronizing shaft for the tilt wing propeller and vectored lift concepts is thus equal to,

$$L = b_{w} - 2d_{1}$$

For the tilting ducted propeller the shaft length was assumed to be equal to the full span, $b = \sqrt{A_R S}$ The shaft length for the "Aerodyne" was assumed to be one diameter plus 10 feet, and for the Breguet-Kappus wherein 6 ducted propellers were assumed, the shaft length is equal to 7 diameters.

The number of propeller synchronizing transmissions are of course governed by the number of propellers which are indicated on the summary charts.

d. Body Weight

This item was derived as for the rotary wing concepts with the exception of determining fuselage wetted area. From preliminary layouts, the fuselage wetted area for each concept was determined. The values for wetted area are included in the summary charts.

e. Wing Weight

Wing weight calculations were based on the previously derived formula. However, for the tilt wing propeller and vectored lift VTOL concepts a constant chord wing was assumed ($\lambda = 1.0$) resulting in a taper factor of 1.4. It should be noted that for these two concepts, the wing span was determined as a function of disc loading and number of propellers. The expression has been derived in Section VI-3C.

f. Miscellaneous

Other items for fixed wing concepts are either a fixed number of pounds or can be expressed directly as a function of gross weight, and therefore do not require further explanation.

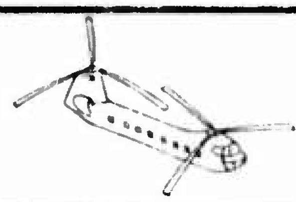
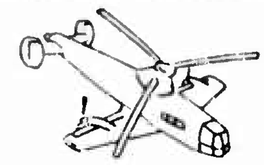
4. Discussion

Obviously the methods employed in this study and the assumptions required to facilitate the numerical calculations, yield results that are not sufficiently accurate to determine, basic design parameters. However, the method used for determining the overall performance of the many VTOL configurations was adopted since it allowed a rapid means of calculation and is considered sufficiently accurate to predict trends for a comparative evaluation.

4. Discussion (Cont'd)

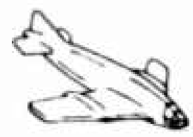
The expressions used for weight trends for performance evaluation are summarized on the following charts.

SUMMARY VTOL WEIGHT TRENDS FOR PERFORMANCE EVALUATION- ROTARY WING CONCEPTS

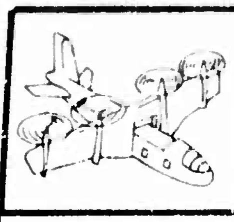
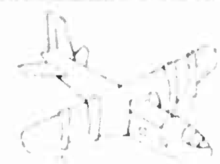
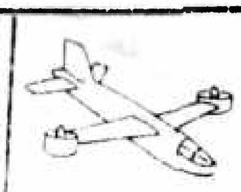
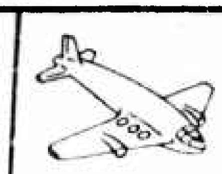
Item	 Tandem Helicopter Shaft Driven Rotors: Conventional Rotor	Tandem Helicopter Shaft Driven Rotors: BLC Rotor	 Compound Helicopter Single Rotor with Propellers: Shaft Driven Rotor – Gas Turbine	Compound Helicopter Single Rotor with Propellers: Shaft Driven Rotor – Turbine Rocket	Compound Helicopter Single Rotor with Propellers: Tip Rocket Driven Rotor
Rotor Group/Rotor Blades	$.0404K^{.63}Wk_w$	$.0445K^{.63}Wk_w$	←	$.0441K^{.63}Wk_w$	→
Hub & Hinge	$.455K^{.53}Wk_w$	$.50K^{.53}Wk_w$	←	$.494K^{.53}Wk_w$	→
Wing Group			← $1.06\left[\frac{156}{C_{L_W}^{.75}\ V^{1.5}}+\frac{.656\,AR^{1.5}}{C_L^{.5}\ V}\right]Wk_w$	→	→
Tail Group	.01 W	.01 W	.029 W	.029 W	.019 W
Body Group	←	$.1807K^{.17}Wk_w$ →	← $.1334K^{.17}Wk_w$	→	$.1219K^{.17}Wk_w$
Alighting Gear*	.04 W	←		→	.04 W
Propulsion Group Rotor	$\frac{.426}{\Lambda_a \pi_t}W$	$\frac{.426}{\Lambda_a \pi_t}W$	$\frac{.516}{\Lambda_a \pi_t'}W$	$\frac{.09\ 6}{\Lambda_a \pi_t'}W + 360$	$\frac{.1456550}{\Lambda_a V_t}W$
Props or Jets				← $.51\ HP_F \times 1.25$	→
Propellers			←	$.2\ HP_F \times 1.25$	→
Drive System Rotor Drive	← $.548K^{.674}Wk_w$	→	←	$.49K^{.674}Wk_w$	→
Prop. Sync. XMSN			$.00897K^{.5}Wk_w$		
Prop. Sync. Shafting			$.832K^{.5}W$		
ϵ	1.04	1.04	1.07	←→	1.07
Number of Nacelles			4		
Number of Rotors	2	2	1	1	1
Correlating G.W.	100,000	100,000	140,000	←→	140,000

* Retractable – Helicopter Design Criteria

SUMMARY VTOL WEIGHT TRENDS FOR PERFORMANCE EVALUATION-ROTARY WING CONCEPTS

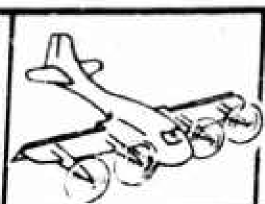
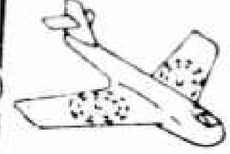
					
Retractoplane Single Rotor with Propellers			Retractoplane Single Rotor with Turbojets		Correlation Factor K
Shaft Driven Rotor		Tip Rocket Driven Rotor	Shaft Driven Rotor Turbine Rocket	Tip Rocket Driven Rotor	
Gas Turbine	Turbine Rocket				
←		$.0529K^{.63}W_{h_w}$		→	$\frac{\epsilon}{(\rho \bar{C}_L V_t^3)10^{-6}}$
←		$.593K^{.53}W_{h_w}$		→	$\frac{1}{\Lambda_a \sqrt{w}}$
←		$1.06\left[\frac{156}{C_L^{.75}V^{1.5}} + \frac{.656AR^{1.5}}{C_L^{.5}V}\right]W_{h}$		→	
.03W	.03W	.019W	.03W	.019W	
← $.143K^{.17}W_{h_w}$	→	$.130K^{.17}W_{h_w}$	$.143K^{.17}W_{h_w}$	$.130K^{.17}W_{h_w}$	$\frac{1}{w}$
.04W	←		→	.04W	
$\frac{.518}{\Lambda_a \pi_r'}W$	$\frac{.096W}{\Lambda_a \pi_r'}+360$	$\frac{.1458550}{\Lambda_a V_r}W$	$\frac{.096W}{\Lambda_a \pi_t'}+360$	$\frac{.1456550}{\Lambda_a V_t}W$	
	← .51HP X1.25	→	← $.277\,T_j$ X 1.25	→	
←	$.2HP_F \times 1.25$	→			
←	$.443K^{.674}W_{h_w}$	.01W	$.443K^{.674}W_{h_w}$	.01W	$\frac{1}{(\Lambda_a V_t \sqrt{w})10^{-3}}$
$.00897K^{.5}W_{h_w}$					$\frac{1}{\Lambda_a}$
$.832K^{.5}W$					$\frac{AR}{V^2 \Lambda_a C_{L_w}}$
1.07	←		→	1.07	
4					
1	1	1	1	1	
140,000	←		→	140,000	

SUMMARY VTOL WEIGHT TRENDS FOR PERFORMANCE EVALUATION-FIXED WING CONCEPTS

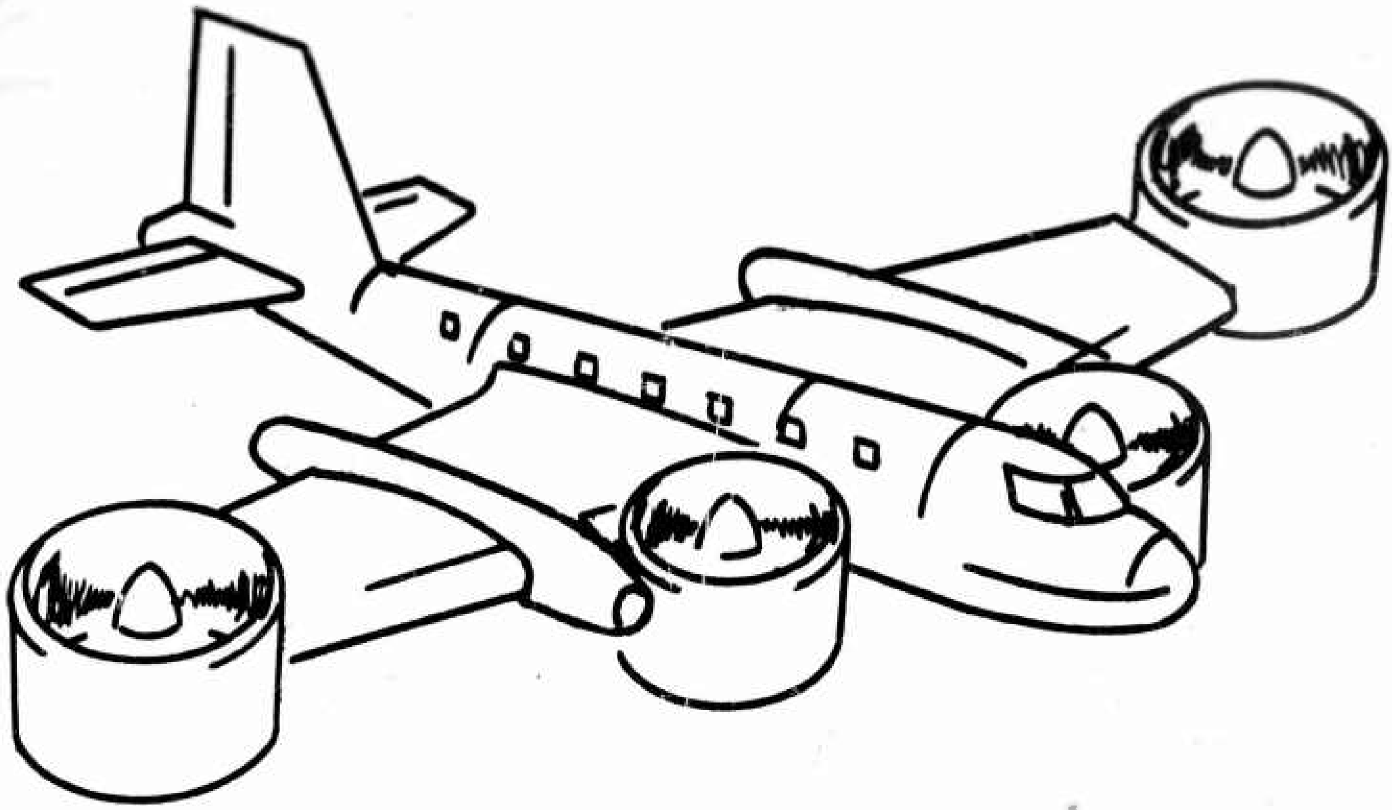
Item	Tilting Wing			Tilting Ducted Propeller	Special Hovering Turbojet
					
	Turboprop	By-Pass Turbojet	Turbojet	Turboprop	
tor Group/Rotor	$27.65 b_p K^{.3062} k_w W$	INCLUDED IN ENGINE WEIGHT			
ng Group	$1.27 k_w W \times \left[K' + \frac{.0004 V^2 C_{L_w} K^3}{b_p \omega^{1.5}}\right]$	$1.27 \times k_w W \left[K' + \frac{.656 AR^{1.5}}{C_{L_w}^{.5} V}\right] k_w W$		$1.165 k_w W \left[K' + \frac{.656 AR^{1.5}}{C_{L_w}^{.5} V}\right]$	SAME AS DEFLECTED THRUST
il Group	.03W	←		→	.03W
dy Group	.0756W	.063W	.063W	.0756W	.063W
ighting Gear*	.045W	.04W	←	→	.04W
opulsion Group Fwd. Flight	$\frac{.516}{\Lambda_a \eta_t} W$	.306 6W	.277 6W	$\frac{.536 W}{\Lambda_a \eta_t}$	.277 8W
Vert. Flight					.12 6W
ropeller/Prop.				$\frac{8W}{\omega} + .586 K k_w W$	
rive System Prop. Sync. XMSN	$.0053 \left(\frac{b_p}{\Lambda_a}\right)^{.5} k_w W$			$\frac{.069 k_w W}{\sqrt{\Lambda_a}\, \omega^{.25}}$	
Prop. Sync. Shafting	$\frac{.0813 K W}{(b_p \omega \Lambda_a)^{.5}}$			$\frac{.059 \sqrt{AR}}{\sqrt{q}\, C_{L_w} \sqrt{\Lambda_a}} W$	
Prop. Extension Shaft	$\frac{.01283 k_w W}{(b_p \omega)^{.5} \Lambda_a}$			$\frac{6.405}{\Lambda_a \omega} k_w W$	
S_F	3000	2400	2400	3000	2400
Number of Propellers	4			4	
Correlating G.W.	100000	140000	140000	100000	140000

* Retractable - Helicopter Design Criteria

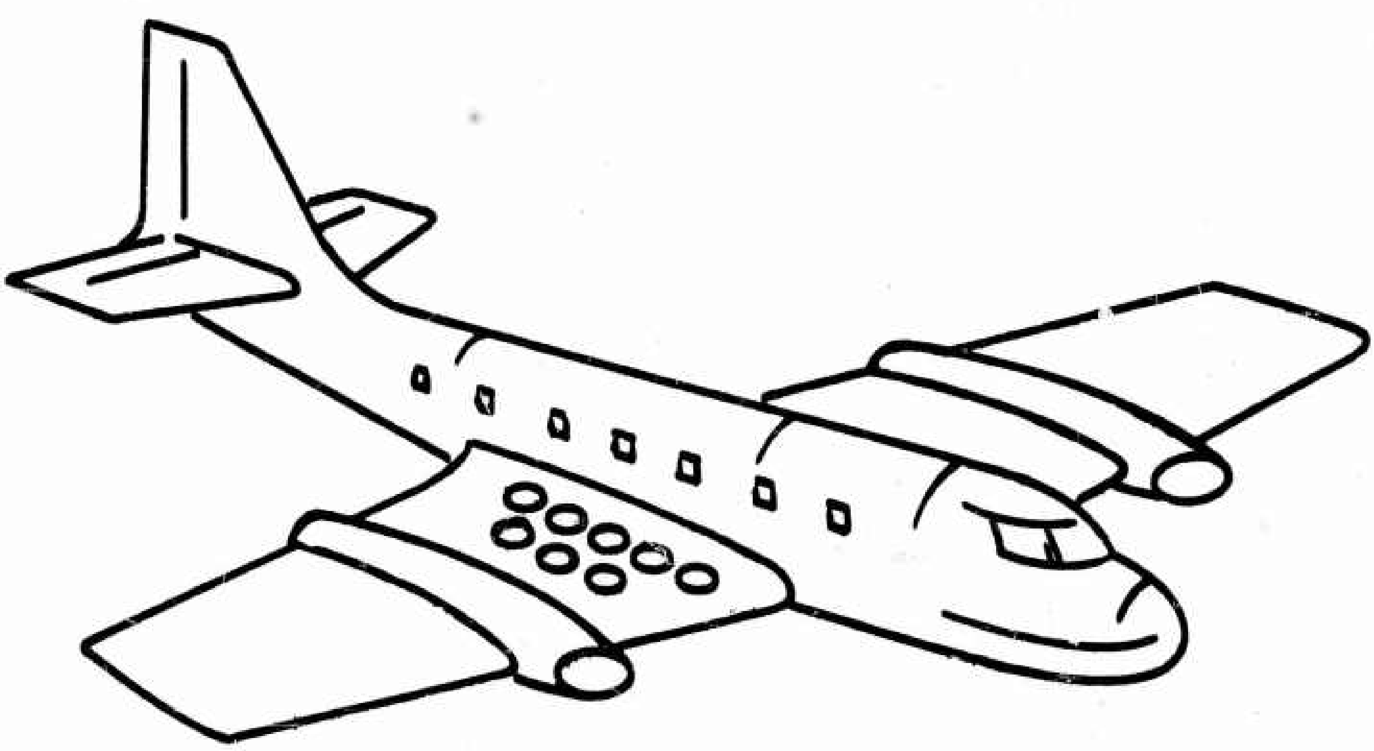
SUMMARY VTOL WEIGHT TRENDS FOR PERFORMANCE EVALUATION-FIXED WING CONCEPT.


Deflected Thrust		Vectored Lift	Brequet-Kappus	Aerodyne	Correlation Factor K
Turbojet	By-pass Turbojet	Turboprop	Split Turboprop	Turboprop	
	INCLUDED IN ENGINE	$27.65 b_p K^{.3062} k_w W$			$\frac{1}{\bar{C}_L w^{1.5} b_p 2 \Lambda_a V_t^2}$
	WEIGHT				
$1.06\left[K' + \frac{.656 AR^{1.5}}{C_L^{.5} V}\right] k_w W$		$1.37 k_w W$ $\left[\frac{.722k^{.75}}{(b_p w)^{.75}} + \frac{.0525K^2}{(b_p w)^{.5}}\right]$	$\left[\frac{.38}{w^{.75}} + \frac{.40}{w^{.5}}\right] k_w W$	$\frac{20}{w} W$	$K' = 156 / C_{L_w}^{.75} V^{1.5}$ $K = .865 b_p + 1.13$
.03W	←		→	.03W	
.063W	.063W	.0756W	.0756	$3W^{.68}$	
.04W	.04W	.05W	.04W	.04W	
.2916 W	.3196 W	$\frac{.516}{\Lambda_a \eta_t} W$	$\frac{.566}{\Lambda_a \eta_t} W$	$.536 HP_{H_A}$	
			$.634 K k_w W$	$5.23\left[\frac{HP_{HA} W^2}{w^{1.5}}\right]^{.27}$	$1/\Lambda_a^{.27} w^{.405}$
		$.0053\left(\frac{b_p}{\Lambda_a}\right)^{.5} k_w W$	$\frac{.0907}{\sqrt{\Lambda_a}\, w^{.25}} k_w W$	$1.189\left[HP_{HA} \sqrt{\frac{W}{\Lambda_a}}\right]^{.5}$	
		$\frac{.0813 KW}{b_p (w \Lambda_a)^{.5}}$	$.039\left(\frac{1}{\eta_t \Lambda_a w}\right)^{.5} W$	$HP_{H_A}^{.5}\left[.574 + .046\left(\frac{W}{w}\right)^{.5}\right]$	$.865 b_p + 1.13$
		$\frac{.01283 k_w W}{(b_p w)^{.5} \Lambda_a}$	$\frac{2.13}{\Lambda_a w} k_w W$	$.00006 \frac{HP_{HA} W}{w}$	
2400	2400	3000	3000	3000	
		4	6	2	
140000	140000	100000	100000	————	



TILTING DUCTED PROPELLER CONCEPT



SPECIAL HOVERING TURBOJET CONCEPT

VII. LIST OF SYMBOLS

AR = aspect ratio = b^2/S

b = wing span, ft.; number of blades per rotor

b_p = number of propellers; number of rotors

C = wing chord, ft., rotor blade chord, ft.

C_1 = non-bending material factor = .024

$\bar{C}_L$ = average rotor lift coefficient = $6w/\rho V_t^2 \sigma$

C_{LW} = operational wing lift coefficient = W/qS

$C_{L_{MAX}}$ = maximum lift coefficient

C_{DO} = profile drag coefficient

C_{Di} = induced drag coefficient

d_1 = propeller slipstream diameter, ft.

D = drag, lbs.; diameter, ft.

e = airplane wing efficiency

f = equivalent flat plate area, sq. ft.; average allowable stress, lbs/sq.in.

HP = horsepower

HP_F = cruise horsepower

HP_H = hover horsepower

HP_P = propeller horsepower

HP_S = horsepower transmitted in the shaft

HP_X = horsepower transmitted in the transmission

k_1 = hovering induced power factor for rotary wing concepts

k_2 = forward flight induced power factor for rotary wing concepts

k_W = gross weight linearization factor

K = weight trend correlation factor

K_R = thickness of wing root chord expressed in percent

K_T = thickness of wing tip root chord expressed in percent

L = range, ft.;length of shaft or fuselage, ft.

[redacted]

[redacted]

$L.F$ = load factor

N = number of transmissions and/or nacelles

q = dynamic pressure lbs/sq.ft. = $\rho V^2/2$

R = rotor or propeller radius, ft.

S = wing area, sq.ft.

S_F = fuselage wetted area, sq.ft.

S_S = shroud surface area, sq.ft.

T_j = thrust, lb.

TF = wing taper factor

v = induced velocity, ft/sec.

V = forward velocity, ft/sec. unless otherwise noted

V_t = rotor or propeller tip speed, ft/sec.

W = gross weight, lbs.

W_B = blade weight, lbs.

W_{CR} = cruise fuel weight, lbs.

W_D = drive system weight, lbs.

$W_{eng.}$ = installed engine weight, lbs.

W_F = body weight, lbs.

W_{FP} = flapping propeller weight, lbs.

W_{FUL} = fixed useful load, as defined in Section III, lbs.

W_{HOV} = hover fuel weight, lbs.

W_L = lift propulsive system weight as defined in Section III, lbs.

W_{LG} = landing gear weight, lbs.

W_P = propeller or propulsive group weight as defined in Section III, lbs.

W_{SX} = synchronizing transmission, lbs.

W_T = tail weight, lbs.

$W_?$ = weight empty less W_P & W_L

β = tankage weight factor

γ' = part-load specific fuel consumption factor

δ = engine power loss factor for 6000' and 95°F operation at static conditions

ϵ = download factor

η_t = rotor transmission efficiency

η_t' = rotor transmission efficiency including anti-torque power loss

η_p = propeller efficiency

λ = equivalent tip chord/root chord

Λ_i = ideal power loading, lbs/HP

Λ_a = actual power loading, lbs/HP

μ = advance ratio, V/V_t; coefficient of friction

π = 3.1416

ρ = mass density, slugs/cu. ft.

σ = rotor solidity = $bcR/\pi R^2$

w = disc loading, lbs/sq.ft. = $W/i\pi R^2$

w_B = blade loading, lbs/sq.ft. = $W/i\sigma\pi R^2$

w_W = wing loading, lbs/sq.ft. = W/S

w_{eng} = specific weight of engine, lbs/HP or lbs/thrust

Ω = propeller or rotor rpm

Ω_S = shaft rpm

Ω_X = transmission rpm (lowest value)

Subscripts

A - refers to Aerodyne

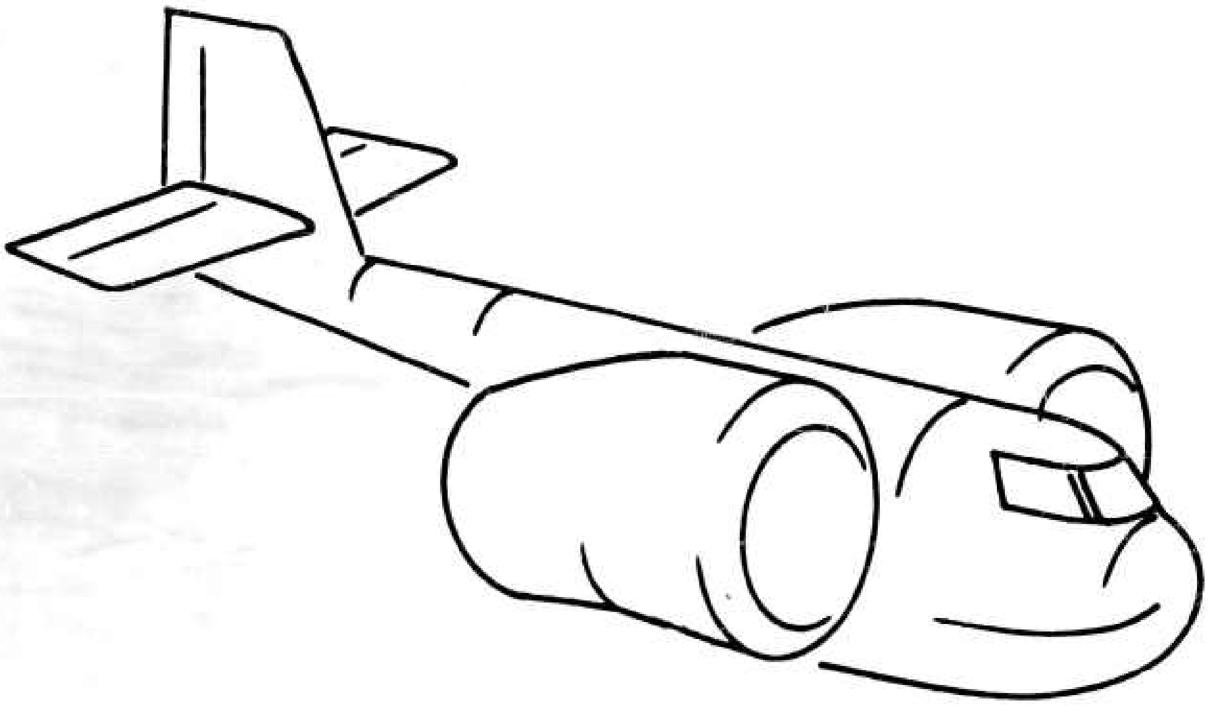
CH - refers to compound helicopter

FW - refers to fixed wing

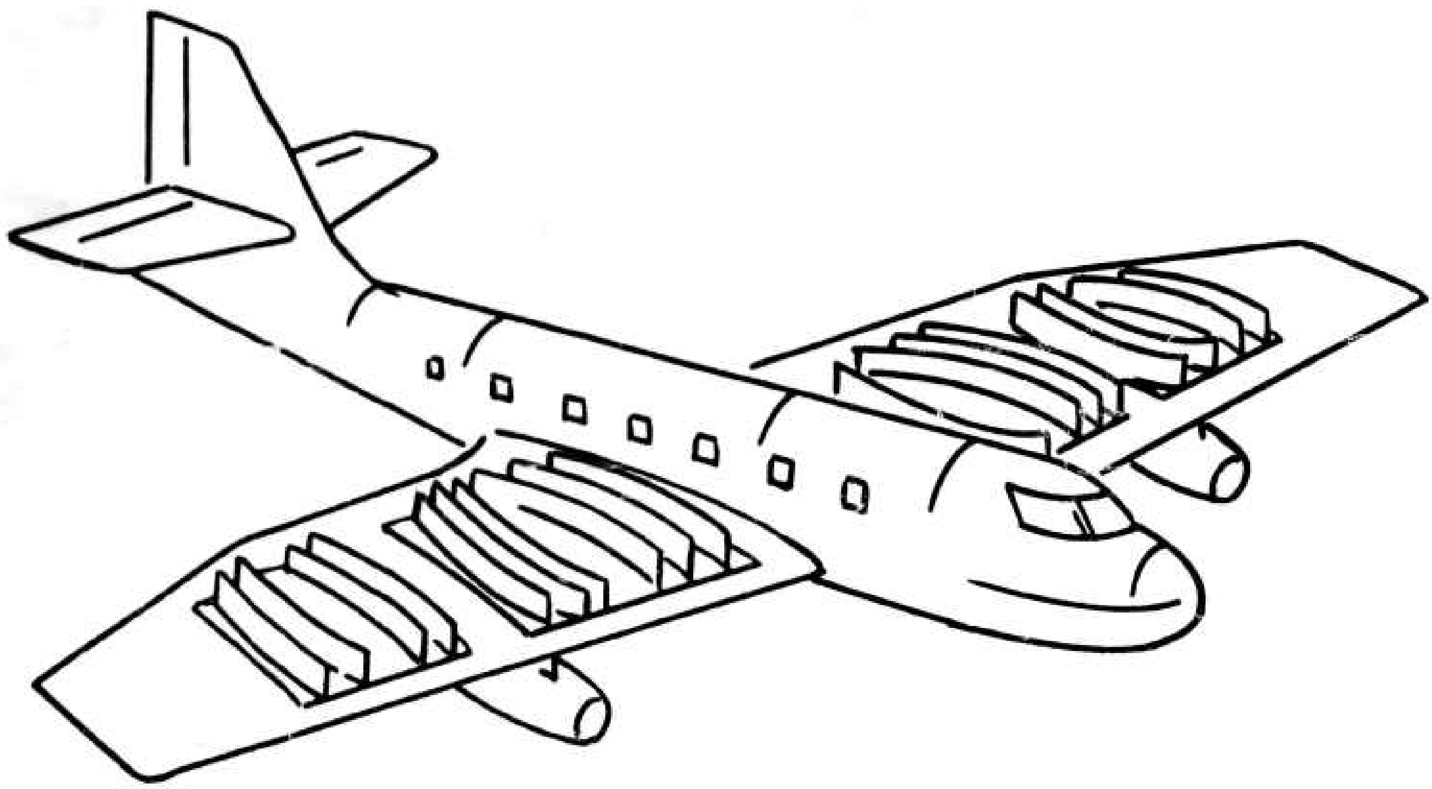
RW - refers to conventional rotary wing

TDP - refers to tilting ducted propeller

W - refers to wing



AERODYNE CONCEPT



BREGUET-KAPPUS CONCEPT

Section VIII - References

1. RMI Letter TPL #6, "Power Units for Vertical Take-off Aircraft", Reaction Motors Incorporated, Denville, N.J., dated August 16, 1955 (Confidential).

2. Stepnewski, W. Z., "Introduction to Helicopter Aerodynamics", Rotorcraft Publishing Committee, Morton, Pennsylvania.

3. Perkins, C.D., and Hage, R.E., "Airplane Performance, Stability and Control", John Wiley and Sons, Inc., 1949.

4. Draper, J.W. and Kuhn, R.E., "Some Effects of Propeller Operation and Location on Ability of a Wing With Plain Flaps to Deflect Propeller Slipstream Downward for Vertical Take-off", NACA Technical Note 3360, January 1955.

5. Draper, J.W. and Kuhn, R.E., "Investigation of the Aerodynamic Characteristics of a Model Wing-Propeller, Combination and of the Wing and Propeller Separately at Angles of Attach Up to 90°," NACA Technical Note 3304, November 1954.

6. Draper, J.W. and Kuhn, R.E., "Investigation of Effectiveness of Large-Chord Slotted Flaps in Deflecting Propeller Slipstream Downward for Vertical Take-off and Low-Speed Flight", NACA Technical Note 3364, January 1955.

7. Rand Report R-269, "Performance Weight and Size Relations for a Family of Turbofan Engines with Exhaust Mixing", the Rand Corporation, Santa Monica, California, June 30, 1954. (Confidential)

8. Engelbry, C.R., "Wing Weight Estimating Simplified", S.A.W.E. Paper No. 20.

9. Platt, R.J., Jr., "Static Tests of a Shrouded and an Unshrouded Propeller", NACA RM No. L7H25, February 9, 1948.

UNCLASSIFIED

UNCLASSIFIED