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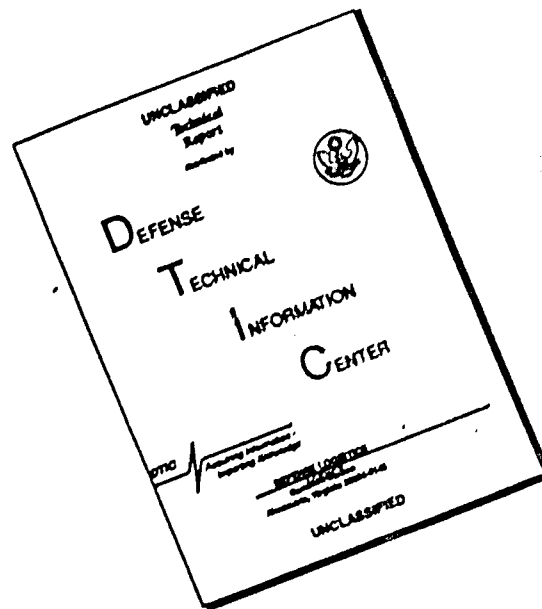
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Office of Naval Research
Contract Nonr 2199 (00)

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DUCTED PROPELLER WEIGHT LIFTER

Advanced Research Division Hiller Helicopters

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BAR PA/100r 2199(00)
GLB:1v
Ser 0196
12 December 1957

FIRST ENDORSEMENT on HILLER HELICOPTERS Conf ltr ARD-57-M127 JBick dtd
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To: Office of Naval Research, Washington 25, D.C. (Attn: Air Branch)

Subj: Contract Noar 2199(00), Ducted Propeller Weight Lifter

1. Forwarded for review and evaluation.

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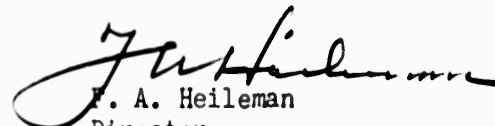
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Subject: Contract Nonr 2199(00). Ducted Propeller
Weight Lifter

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HILLER HELICOPTERS


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ONR WEIGHT LIFTER

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A Technical Study for
The Office of Naval Research,
Department of the Navy

A DUCTED PROPELLER WEIGHT LIFTER

Office of Naval Research
Contract Nonr 2199(00)

November 22, 1957

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Report No. 149

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NOTE

Assumptions and limiting factors pertaining especially to ducted propellers are indicated by underlines, per ONR letter ONR:461:TLW:ecw of 1 October 1957.

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LIST OF SYMBOLS

- A = Propeller disk area, ft^2
- A_e = Flow area through duct, ft^2
- A_t = Area of vertical tail, ft^2
- $\frac{A_4}{A_2}$ = The ratio of slipstream area downstream to that of the propeller
= 1.0 for ducted propellers with straight existing ducting
- A^* = Activity factor
- B = Width of fuselage, ft
- B_1 = Number of blades
- B_D = Thickness of duct, ft
- b = Number of ducts
- b_1 = Wing span, ft
- C_{D_e} = External drag coefficient, based on disk area
- C_{D_i} = Internal drag coefficient of ducted propeller, based on disk area
- CG = Center of gravity
- C_L = Coefficient of lift
- C_{L_R} = Mean blade lift coefficient
- D = Diameter of propeller, ft
- D_e = External drag = $C_{D_e} \frac{\rho V_o^2}{2} A$, lb
- D_i = Internal drag of ducted propeller = $C_{D_i} \frac{\rho V_o^2}{2} A$, lb
- DP = Ducted propeller

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- f_1 = Design stress factor
- H = Height of fuselage, ft
- H_D = Height of duct, ft
- HP = Horsepower
- K = Empirical constant
- L = Length of fuselage, ft
- L_D = Circumferential length of duct, ft
- $LP_m = \frac{W_G}{1.167HP}$, lb/HP
- LF = Level flight
- M = Figure of merit for static thrust
- M_D = Pitching moment of duct, ft-lb
- M_F = Pitching moment of fuselage, ft-lb
- M_R = Pitching moment due to residual thrust, ft-lb
- M_T = Pitching moment due to thrust-CG misalignment, ft-lb
- m = Mass flow per second, lb-sec/ft
- N = rpm
- n = Number of installed engines
- P = Payload, lb
- PP = Pusher propeller
- Q = Output shaft torque, ft-lb
- q = Dynamic pressure, lb/ft²
- R = Range, nautical miles
- $R_c = \text{Crew weight to gross weight ratio} = \frac{W_C}{W_G}$

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R_F = Fuel weight to gross weight ratio = $\frac{W_F}{W_G}$

T = Thrust, lb

T_J = Residual thrust from engine, lb

V_a = Aircraft speed, knots

V_e = Duct exit velocity, ft/sec

V_o = Free-stream velocity, ft/sec

V_T = Tip speed of propeller, ft/sec

V_2 = Downwash velocity = $\left(\frac{w_e}{\rho}\right)^{\frac{1}{2}}$, ft/sec

W_B = Fuselage weight, lb

W_C = Crew weight, lb

W_D = Duct weight, lb

W_E = Engine weight, lb

W_F = Fuel weight, lb

W_G = Gross weight, lb

W_g = Gear box weight, lb

W_L = Landing gear weight, lb

W_m = Miscellaneous items weight, lb

W_O = Oil and oil tank weight, lb

W_P = Powerplant and accessories weight, lb

W_R = Rotor weight, lb

W_S = Shaft weight, lb

W_T = Transmission weight, lb

W_W = Wing weight, lb

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- W_{BP} = Tail boom and exhaust pipes weight, lb
- W_{DS} = Duct support structure weight, lb
- W_{EX} = Empennage weight, lb
- W_{ES} = Engine section weight, lb
- W_{FE} = Fixed equipment weight, lb
- W_{FT} = Fuel tank weight, lb
- W_{HS} = Horizontal tail weight, lb
- W_{SS} = Starting system weight, lb
- W_{VT} = Vertical tail weight, lb
- W_{TM} = Tilting mechanism weight, lb
- $W_{empty} = W_G - (W_C + P + W_F + W_{FT}), lb$
- w = Disk loading, based on propeller disk area, lb/ft^2
- w_e = Effective disk loading, based on flow area, lb/ft^2
- c = Angle of forward tilt of ducted propeller, degrees
- γ = Forward tilt angle of the ducted propeller thrust vector, degrees
- δ = Angle between the horizontal and the resultant ducted propeller slipstream, degrees
- ϵ = Velocity ratio = $\frac{V_e}{V_o}$
- η_p = Propulsion efficiency
- η_t = Transmission efficiency
- η_{tail} = Efficiency of horizontal tail
- ρ = Density of air, $lb\text{-}sec^2/ft^4$
- σ = Air density ratio = $\frac{\rho}{\rho_o}$

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τ = Ducted propeller efficiency factor for forward flight =

$$\frac{(2)^{.5} (\gamma)^{1.5}}{e(e^2-1)}$$

ϕ = $\frac{\text{Component Weights}}{\text{Gross Weight}}$

γ = Ducted propeller non-dimensional lift parameter = $\frac{W_0/\lambda_0}{\rho V_0^2}$

$l \cdot f$ = Flow area • equivalent flat plate area of drag surfaces, $\text{ft}^2 = 1.25$

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1. SUMMARY

Hiller Helicopters received Contract Nonr 2199(00) from the Office of Naval Research to conduct theoretical and preliminary design studies of the application of ducted propellers as direct lift devices for a specialized wight-lifting vehicle. Work on this contract was begun on 15 October 1956, to be completed one year from that date. After several months of investigation, it was determined that to refine these studies would be unjustified and possibly misleading, in light of the gross assumptions necessary concerning the forces and moments produced by a ducted propeller. For this reason the original program was concluded 1 June 1957.

The aircraft design requirements used for the Weight Lifter study were specified as follows:

- a. Vertical take-off and landing capability at a sea-level standard day.
- b. Ability to hover at a sea-level standard day with satisfactory control.
- c. Installed horsepower 20% greater than that necessary to hover, to allow for power needed to climb.
- d. Powerplant and propeller specifications as expected by 1962.
- e. A minimum forward cruise speed of 50 to 70 knots.
- f. A range of aircraft sizes to carry from 7 to 16 tons payload.

A survey of previous studies indicated that ducted propellers are considered to be a quite satisfactory means of powering a vertical take-off weight-lifter type aircraft, as their static thrust is considerably higher than for unshrouded propellers, thereby reducing the required horsepower. Also, the Weight Lifter uses conventional components and construction methods wherever possible, which contributes to lower cost and increased reliability. An example of this is the use of conventional variable pitch propellers.

As no mission was specified, all calculations were made on the basis of the hovering requirement, and the maximum forward flight speed of 150 knots was the result of the horsepower installed to hover at an optimized disk loading. The lack of theoretical or empirical data for ducted propellers in forward flight has necessitated the use of calculations based on momentum theory to determine the performance of the aircraft in this study.

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This study began with an analysis of five basic configurations, described briefly below and shown in Figures 1 through 5.

- a. Two ducted propellers, each mounted at the tip of a straight wing of a conventional transport type aircraft.
- b. Two ducted propellers mounted on the fuselage between two wings, similar to a canard design.
- c. Three ducted propellers built into a delta flying wing.
- d. Two ducted propellers mounted on the fuselage, replacing wings of a conventional transport type aircraft.
- e. Four ducted propellers mounted on a fuselage, replacing wings and tail surfaces.

As always, the cost of an aircraft is directly associated with its weight; therefore, a weight criterion for choosing a configuration in the preliminary design stage is necessary. In this study, an adaptation of the R_p Graphical Method of Reference 10 is used, translating all weight values to a ratio of fuel weight to gross weight, R_p , for direct comparison. The four basic parameters of this study are gross weight, number of ducts, payload, and disk loading.

For the conventional components and equipment of the aircraft, suitable weight equations were adopted from previous studies. If existing equations did not apply, an adjustment was made by actually designing the unconventional component and developing a new weight equation to suit the provisional design.

The Weight Lifter preliminary design was determined as a result of an analysis of the five configurations listed above and a subsequent optimization of various parameters for the selected configuration. The optimization was based on a criterion of maximum payload ton-mile per pound gross weight. The configuration that emerged as best is shown in the Frontispiece.

As the study progressed, it became apparent that internal cargo stowage would be more satisfactory than external loading. It is now believed that the weight penalty for a large fuselage would not be as severe as anticipated. This conclusion was reached after considering that special cargo containers would be necessary to protect many of the types of cargo from weather and the strong propeller downwash and its accompanying flying debris during take-off and landing. Also, the flight speeds and ranges possible with the final configuration are largely dependent upon a clean design.

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The ducted propellers are positioned horizontally to provide direct lift for hovering and vertical flight, and may be tilted forward 55 degrees to provide forward propulsion as well. The counter-rotating propellers are powered through extension drive shafts by several interconnected turboshaft engines mounted on the top of the fuselage. Only turboshaft powerplants have been considered in this study, due to their high horsepower-to-weight ratio. While no specific engine make has been assumed, representative weights and performance values have been used.

The residual thrust of the turboshaft engines is ducted through the fuselage boom to a thrust-diverting nozzle at the aft of the aircraft, and provides the pitch and yaw control forces in hovering and low-speed flight. The tail surfaces furnish pitch and yaw control in forward flight, and the residual thrust is then utilized to supplement the forward propulsive force of the tilted ducted propellers. Roll control is supplied by differential thrust of the ducted propellers, both in hovering and forward flight.

The Weight Lifter design has been optimized at various overall sizes to accommodate the specified range of payload, as shown in the following table. Optimum disk loading was found to be 150 pounds per square foot for all values of gross weight.

Payload	7 tons	16 tons
Gross Weight	60,400 lb	135,000 lb
Power Required	18,200 HP	40,800 HP
Ducted Propeller Dia.	18 ft	26.7 ft
Range	180 n.mi.	135 n.mi.
Max. Velocity	150 knots	150 knots

The three-view drawing seen in Figure 19 is of a representative aircraft, designed to accommodate an average payload of 11 tons.

A lack of certain theoretical and experimental information on ducted propellers has necessitated gross assumptions in the calculations involved in this analysis. In order to refine this study, a follow-on ducted propeller wind tunnel test program has been suggested by this contractor.

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2. INTRODUCTION

Hiller Helicopters received Contract Nonr 2199(00) from the Office of Naval Research to conduct theoretical and preliminary design studies of the application of ducted propellers as direct lift devices for a specialized weight lifting vehicle. Work on this contract was begun on 15 October 1956, to be completed one year from that date. After several months of investigation, it was determined that to refine these studies would be unjustified and possibly misleading, in light of the gross assumptions necessary concerning the forces and moments produced by a ducted propeller. For this reason, the original program was concluded on 1 June 1957.

This study has consisted of theoretical and analytical investigations of several possible Weight Lifter configurations. The configurations which appeared most satisfactory were selected for further design studies to optimize the physical, performance, and control characteristics. This report reviews the investigations and design philosophies which led to the final configuration selection, and summarizes the results of the study program. Performance figures are presented for both the ducted propeller and the complete aircraft. Weights and important details of the structural design are also included.

The aircraft design requirements used for the Weight Lifter study were specified as follows:

- a. Vertical take-off and landing capability at a sea-level standard day.
- b. Ability to hover at a sea-level standard day with satisfactory control.
- c. Installed horsepower 20% greater than that necessary to hover, to allow for power needed to climb.
- d. Powerplant and propeller specifications as expected by 1962.
- e. A minimum forward cruise speed of 50 to 70 knots.
- f. A range of aircraft sizes to carry from 7 to 16 tons payload.

As the study progressed, it became apparent that internal cargo stowage would be more satisfactory than external loading. This is due in some measure to the strong downwash associated with the high disk loadings called for in this study. Also, the flight speeds and ranges possible with the final configuration are largely dependent upon a clean design. Provision may be made, however, for emergency external loading of very bulky articles which need to be moved only short distances at low speeds, and for such occasions when cargo must be loaded or unloaded without landing.

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It is now believed that the weight penalty for a large fuselage would not be as severe as anticipated. This conclusion was reached after considering that special cargo containers would be necessary to protect many of the types of cargo from weather and the strong propeller downwash and its accompanying flying debris during take-off and landing. This is especially true of troops and assembled equipment containing glass, light metals, or fabric. A separate cargo container of this type would not contribute to the load-bearing structure of the aircraft, therefore, the total effective weight of the cargo container and airframe may well be as great as that of the internally loaded aircraft.

Only turboshaft powerplants have been considered in this study due to their high horsepower-to-weight ratio. While no specific turboshaft engines have been assumed, representative weights and performance values have been used.

The location of the engines has a great deal to do with the weight and the configuration of the airframe; therefore, the safety aircraft performance, control, and reliability of various engine locations have been considered in determining the optimum Weight Lifter.

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3. APPROACH TO THE PROBLEM

3.1 Ducted Propeller Characteristics

The basic design problem of this study was the investigation of a system of ducted propellers mounted about a cargo carrying fuselage. A survey of previous studies (References 1, 2, and 3) has indicated that ducted propellers are considered to be a quite satisfactory means of powering a vertical take-off weight-lifter type aircraft. The limited test data available on ducted propellers indicate that their static thrust is considerably higher than for unshrouded propellers, thereby reducing the horsepower required to hover. Ducted propeller type aircraft offer another advantage in that they utilize components which have been previously developed under extensive fixed wing aircraft development programs, and therefore display relatively low flight hour cost per pound gross weight (Reference 1).

The lack of theoretical or empirical data for inclined ducted propellers in forward flight has necessitated the use of calculations based on aerodynamic theory to determine the performance of the aircraft in this study. Ducted propeller pitching moments were obtained largely from the Hiller Flying Platform track tests (Reference 4).

3.2 Work Statement

The basic designs investigated in this study are briefly described as follows

- a. Two ducted propellers, each mounted at the tip of a straight wing of a conventional transport type aircraft.
- b. Two ducted propellers mounted on the fuselage between two wings; similar to a canard design.
- c. Three ducted propellers built into a delta flying wing.
- d. Two ducted propellers mounted on the fuselage, replacing wings of a conventional transport type aircraft.
- e. Four ducted propellers mounted on a fuselage, replacing wings and tail surfaces.

Plan view and side view drawings of these configurations are shown in Figures 1 through 5.

Preliminary design studies have been carried out to the extent that both hovering and forward flight characteristics of each configuration were calculated. The following major design areas were investigated:

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- a. Optimum propulsion system
 - 1. Type and location of powerplants
 - 2. Disk loading of the selected propeller
- b. Total power required
 - 1. Lift
 - 2. Propulsion
 - 3. Control forces
- c. Control moments required
 - 1. Hovering
 - 2. Forward flight

3.3 Design Procedure

As no combat or transportation mission was specified for the Weight Lifter study, the basic design parameters were established about hovering conditions with full payload at a sea level standard day for a range of disk loadings. The horsepower required to hover at these conditions was increased by 10% to provide for power to climb. The gross weight and horsepower required per pound of payload were calculated and a general configuration determined for the five Weight Lifter designs through a range of disk loadings, as discussed in Section 3.1. below. At this point, three of these designs were eliminated from further investigation by a preliminary analysis and comparison of gross weights, complexity, and general applicability to the Weight Lifter type of service. Only those configurations showing the most promise were investigated further.

The horsepower required for hovering plus climb was then used to compute the maximum forward velocity of the aircraft in level flight. This procedure resulted in defining two distinct types of aircraft: one, a very efficient hovering machine with a low disk loading, low speed, capable of only short range, and somewhat lighter than the other type, which is an aircraft requiring greater horsepower to hover but has greatly increased speed and range capabilities. An operations research analysis (Reference 1) indicated that the higher speed, longer range design will perform the anticipated Weight Lifter missions much more satisfactorily than the low speed short range design if the weight increase accompanying the increased performance is not too great. For this reason, further design studies were performed only for the higher speed aircraft, for which a break-even point was established between hovering efficiency and range to determine the optimum aircraft for general service. Stability and control values were then determined for the configuration designated by the performance optimization studies.

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3.4 Performance Calculations

3.4.1 Hovering

The theoretical expression for the thrust to power ratio required for hovering can be expressed as

$$\frac{\text{Thrust}}{\text{Power}} = M \sqrt{\frac{2\rho}{\gamma A}}$$

where M is defined as a figure of merit or hovering efficiency of the propeller. It should be noted that the propeller efficiency as defined by the preceding equation originally refers to an unshrouded propeller for which the ideal theoretical value is $M = 1$. For convenience, the same definition is also used for the shrouded propeller. In this case, however, the ideal theoretical value becomes $M = \sqrt{\gamma}$ which explains the perhaps surprising fact that the actual measured efficiencies are larger than unity. If η_t is the transmission efficiency, the horsepower required to hover then becomes

$$(\text{HP})_{\text{hovering}} = \frac{W_G}{550M\eta_t} \sqrt{\frac{W_G}{\gamma}}$$

A survey of the available static thrust data for ducted propellers was conducted in order to obtain a realistic value of figure of merit for a ducted propeller of the type anticipated for the Weight Lifter. As stated in Reference 5, the maximum figure of merit of the ducted propellers tested by Krüger (Reference 6) was approximately 1.15. It should be noted, however, that this propeller-shroud combination was designed for an advance ratio of 0.95. It may be expected that higher figures of merit can be obtained by using a duct form which favors the low speed range, possibly one which incorporates duct inlet flaps (References 3 and 6). The data of Reference 7 tends to substantiate this assumption by indicating a figure of merit of 1.5 for a "short-cruise" shroud. Other test data analyzed in Reference 5 also indicate a similar range of values for figure of merit.

From this survey it was assumed that a counterrotating ducted propeller of a design suitable for a Weight Lifter type aircraft could produce a figure of merit, including transmission losses, of

$$M\eta_t = 1.28^*$$

(*Note. In the interim between the survey of the figure of merit described above and the publication of this report, further studies, including a conference with the NACA (Reference 8), indicate that the maximum value of figure of merit for counterrotating ducted propellers may be of the order of

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$M = 1.20$

As the direct relationship between the figure of merit and the horsepower required to hover is the same for each of the configurations investigated it may be seen that altering the figure of merit would not affect the selection of the basic weight lifter configuration. A revision of the representative optimum Weight Lifter (Figure 19) on the basis of the lower figure of merit resulted in an increase of .0% in the horsepower required to hover, from the equations of Section 5. This results in a corresponding increase of approximately 1% in the gross weight of the aircraft. Assuming that the increased weight of the powerplants and related components is counteracted by reducing the fuel and fuel tank weight to maintain the original gross weight, the range of the aircraft will decrease approximately 1%. From this examination it may be seen that if the lower figure of merit were used it would not greatly affect the characteristics of the Weight Lifter.)

The horsepower required in hovering was calculated for a range of aircraft gross weight and lift loading, and plotted as a curve graph in Figure 6. The horsepower required to hover for each case was increased by .0% to allow for sufficient power to climb.

3.3.1 Forward Flight

The performance calculations for level flight have been based upon equations derived from momentum theory. This method neglected compressibility effects and interference effects between ducted propellers in close proximity to one another.

Three methods of obtaining a component of thrust to provide forward propulsion to the aircraft were investigated. Method 1 consists of tilting the ducted propellers forward, the forward component of the thrust vector provides all of the propulsive force. Method 2 allows the ducts to tilt forward only enough to overcome their own momentum drag, additional propulsive force comes from pusher propellers. Method 3 has the ducted propellers remain fixed in the hovering position, all propulsive force is supplied by pusher propellers.

As the study progressed, the use of lifting surfaces in addition to the ducted propeller were found not to be practical and Method 1, propulsion only from the tilted ducts, resolved itself into a force system depicted by the vector diagram shown in Figure 7. For this method, the thrust vector must provide all lifting force and propulsive force components. From the vector diagram and momentum theory,

$$T^2 = (D_e + D_i \sin \alpha)^2 + (W_G + D_i \cos \alpha)^2 - m^2 V_o^2 + m^2 V_e^2 - 2m^2 V_o V_e \sin \alpha$$

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This equation can be reduced to

$$\epsilon^4 (1 - 0.5C_{D1})^2 - \epsilon^2 - \epsilon C_{D0} - 1^2 + 0.25C_{D0}^2$$

by introducing the nondimensional coefficients

$$\epsilon = \frac{V_e}{V_0}$$

and

$$\gamma = \frac{W_G / A_0}{\rho V_0^2}$$

The coefficient γ corresponds to the lift coefficient of conventional lifting surfaces. The above equation permits the calculation of the duct exit to free stream velocity ratio, ϵ , as a function of disk loading, free stream velocity, external and internal drag.

From momentum theory

$$(\text{power})_{\text{ideal}} = \pi/2 (V_e^2 - V_0^2)$$

If η_p and η_t are the propulsive and transmission efficiency respectively, horsepower required for level flight becomes

$$(\text{HP})_{\text{LF}} = \left(\frac{W_G V_0}{2(550) \eta_p \eta_t} \right) \left(\frac{\epsilon(\epsilon^2 - 1)}{\gamma} \right)$$

This equation can be expressed similarly to that for hovering by the introduction of a nondimensional value, τ , which represents a figure of merit for forward flight. Horsepower required for level flight then becomes

$$(\text{HP})_{\text{LF}} = \frac{W_G}{550 \tau \eta_p \eta_t} \sqrt{\frac{W_e}{2\rho}}$$

where

$$\tau = \frac{\sqrt{2}(\gamma)^{1.5}}{\epsilon(\epsilon^2 - 1)}$$

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A carpet plot of HP required for level flight is shown for Method 1 as a function of disk loading and forward velocity in Figure 8. For this study, $\eta_p \eta_t$ has been assumed to be 0.85.

The second method of providing forward propulsion described above has a similar derivation as shown for Method 1 and produces the expression

$$\epsilon^2(1 - 0.5C_{Di})^2 = \epsilon^2 - Y^2$$

The horsepower required for the ducted propeller in level flight is

$$(HP)_{LPP} = \left(\frac{W_0 Y_0}{(550) \eta_p \eta_t} \right) \left(\frac{\epsilon(\epsilon^2 - 1)}{1} \right)$$

The horsepower required for the pusher propeller is

$$(HP)_{LPP} = \left(\frac{W_0 Y_0}{(550) \eta_p \eta_t} \right) \left(\frac{C_{De}}{Y} \right)$$

The forward tilt angle necessary for the ducted propeller to overcome its own momentum drag is given by the expression

$$\tan \alpha = \frac{\epsilon}{Y}$$

It may be seen that the second method has no advantage over the first, as the rearward vector of the momentum change through the ducted propellers contributes the greatest portion of the aircraft drag. The ducts would have to be tilted forward very nearly the same amount to overcome the duct momentum drag as for the total aircraft drag. The calculations comparing the two methods show nearly identical values for horsepower required, and the extra weight and complexity of the powerplant and drive system for the pusher propellers would be a disadvantage when comparing methods.

The third method proposed, which would provide all forward propulsion by pusher propellers, yields the expression

$$\epsilon = \sqrt{\frac{Y}{1 - 0.5C_{Di}}}$$

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The horsepower required for the ducted propeller is expressed by

$$(HP)_{L_{DP}} = \left(\frac{W_G \gamma_0}{2(550) \eta_p \eta_t} \right) \left(\frac{C(e^2 - 1)}{1} \right)$$

The horsepower required for the pusher propeller is

$$(HP)_{L_{PP}} = \left(\frac{W_G \gamma_0}{2(550) \eta_p \eta_t} \right) \left(\frac{2e \cdot C_{D_e}}{1} \right)$$

The total horsepower required was calculated for this method using the same values as those used for Methods 1 and 2 described above. A carpet graph of total horsepower required was plotted as a function of disk loading and forward speed in Figure 9.

A comparison between Method 1, which tilts the ducts for forward propulsion, and Method 3, which uses only pusher propellers for forward propulsion, shows the great advantage of tilting the ducted propellers for forward flight. At the higher disk loadings and higher speeds, the fixed ducts and pusher propeller combination requires 50% to 80% more horsepower than the tilting duct design.

An investigation of ducted propeller shroud profile shapes and chord lengths (Reference 6) indicated that a short-chord, thin airfoil section shroud was the most efficient for forward flight. For hovering conditions, a large radius or bell-mouth inlet is the most efficient. Recent tests (Reference 9) show that the effects of duct inlet configuration on performance at high forward speeds are less critical at high disk loadings. From these considerations, an average thickness profile shape was assumed for the purposes of this study. Such a duct could be supplemented in hovering with inlet flaps or a pneumatic leading edge. The data of Reference 6 indicated little change in performance for shroud chords as short as 18% of the propeller diameter. For this study, a chord length of 0.18D was used for calculating aircraft performance and duct weights.

Evaluation of test data and preliminary numerical studies indicate that values of duct internal drag coefficient, C_{D_i} , of approximately 0.080 to 0.095 must be expected. The calculations of this study have been based upon

$$\underline{C_{D_i} = 0.1}$$

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Due to the lack of data on the drag of ducted propellers at angles of attack, the flat plate area drag of the ducts in the hovering position was used as a conservative approximation of external drag for all performance calculations. To this value was added the drag of the fuselage, duct supports, and tail sections. The additional drag was found to be such a small part of that estimated for the ducts that it was not changed with slight changes in aircraft configuration. The external drag coefficient, C_{D_e} , was then taken as

$$C_{D_e} = \frac{5(5)(0.155)}{576} = 0.0064$$

based upon the total disk area of the ducted propellers.

3.3.3 Method of Utilizing Disk Loading and Gross Weight

The Hillier Helicopters R_F graphical method of parametric analysis for an optimum aircraft preliminary design (Reference 10) was originally used to determine the optimum Weight Lifter configuration based upon hovering conditions. This method defines the "best design" as the minimum gross weight to perform any specific mission. As no mission was specified for this study, the hovering requirements were used as the parameters for analysis to determine the optimum aircraft. The ratio of fuel weight to gross weight, designated R_F , is used as the basic line through which aerodynamic and weight equations are solved simultaneously by a graphical intersection method. Plots of this graphical solution are shown in Figures 10 and 11. Examination of these curves reveals the optimum points at relatively low disk loadings. These disk loadings and gross weights produce rather low speeds and short ranges, as determined from the carpet plots of horsepower required for forward flight (Figures 8 and 9).

As a result of the previously mentioned operations research investigations and the lack of a specified mission radius of action, a criterion based on tons of payload, flight distance, and gross weight was established as

$$\frac{\text{ton-n.mi}}{\text{lb}} = \text{max.}$$

To accomplish this optimization, the flight distance (range) to gross weight ratio, R/W_G , was plotted as a function of time of flight for a range of payloads and disk loadings from the R_F vs gross weight curves.

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The maximum range to gross weight ratio was determined for each disk loading and payload. The gross weights corresponding to the maximum R/W_0 ratios were plotted in carpet form as a function of disk loading and payload, as shown in Figure 12. From this plot, the disk loading corresponding to the lowest gross weight for any payload can be determined. For a given payload, this disk loading is then the design point which provides the lowest gross weight corresponding to a maximum range to gross weight ratio.

The optimum disk loading and gross weight values were then used to determine the configuration dimensions and performance.

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II. DESIGN STUDIES

II.1 Configuration Investigations

The first configuration study was undertaken for a design which mounted two ducted propellers, one at each wingtip, on a rather conventional transport type aircraft, as shown in Figure 1.

The horsepower required for hovering was calculated as discussed in Section 3.1.1, and plotted in carpet form as a function of gross weight and disk loading (Figure 6). This graph is applicable to all configurations if it is assumed that varying the number of ducts necessary to achieve a given total disk area does not affect the overall figure of merit or the transmission efficiency of that total disk area.

The investigation conducted to determine wing size, weight, and performance indicated that any wing of reasonable size, even equipped with high lift devices, would begin to support the major portion of the aircraft weight only when speeds are reached which comprise the upper extremity of a speed spectrum compatible with a Weight Lifter type operation. The use of a wing on a Weight Lifter, then, presents a paradox. The addition of a wing to the aircraft increases the gross weight and consequently, the horsepower required to hover. Once the aircraft has accelerated to a velocity where the wings are capable of supporting the full load, the additional horsepower installed to hover is then available to continue accelerating to a much higher velocity. The winged configuration thus lends itself to the assault transport type aircraft with its higher speeds and long ranges, rather than to a Weight Lifter where hovering and low speed forward flight constitutes a large part of the total flight time.

Another winged design was investigated in which two ducted propellers were mounted on either side of the fuselage between two wings in a canard configuration, as shown in Figure 2. The result of this study was less satisfactory than for the wingtip-mounted ducted propellers described above, as the two smaller wings would require a larger total area than the single wing to produce the same lift. This design has an additional disadvantage, inherent in canard configurations, in that the rear wing is in the influence of the front wing. Even more significant in this case would be the effect of the inlet and exit flows of the ducted propellers on the wings, as well as the effect of the front wing on the duct inlet conditions.

A delta shaped flying wing with three ducted propellers imbedded in the wing, as shown in Figure 3, was analyzed and found rather poorly suited for Weight Lifter operations. Lower lift coefficients can be expected from this shape wing; therefore, larger areas or even higher speeds would be necessary to support the aircraft weight. The ducted propeller

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inlet conditions would be expected to be very poor, due to the fact that the duct inlet location on the upper surface of the wing would be in a low pressure region when the wing was producing lift during forward flight.

The fixed ducts of this configuration necessitate the addition of pusher propellers to provide forward propulsion. As stated previously, fixed ducted propellers in conjunction with pusher propellers require 50% to 60% more horsepower than tilting ducted propellers at the same disk loading and forward flight speed. A weight analysis indicated that the delta wing would be a much heavier aircraft than the other configurations investigated, because of the large wing area necessary to obtain a sufficient cargo volume and lift area, as well as the added weight of the pusher propeller system.

A configuration was investigated which consisted of two fixed propellers mounted on a fuselage, resembling the wings of a transport type aircraft (see Figure 4). The fixed propellers in this case provide both the lifting and propulsive force. The propeller axis is in a vertical position during the hovering flight of the aircraft, and can tilt forward to provide a thrust component for forward flight. The fuselage remains in a horizontal position for all flight conditions.

The horsepower required for hovering is somewhat less for this configuration, because the gross weight is reduced by the amount of the various wing weights and pusher propeller systems. It was found that the horsepower required for vertical take-off was also sufficient to provide a reasonable forward flight speed; maximum velocity proved to be a function of the disk loading involved.

The final configuration investigated for this study was a design incorporating four ducted propellers as shown in Figure 5. This design also requires tilting the ducted propellers to provide forward propulsion as well as the lifting force, while the fuselage remains horizontal. The front and rear sets of ducted propellers were assumed to have the same dimensions.

The weight of this configuration was calculated and found to be slightly higher than the two-duct no wing design. This was due to the greater duct circumference necessary to encompass the required disk area, the redundancy of the propeller drive systems and tilting mechanism, and the increased fuselage structure necessary to support the four ducted propellers.

The four-duct design has the advantage of not requiring reaction controls for hovering, nor control surfaces during forward flight. All control moments are generated by differential control of the four ducted propellers.

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... Comparison of Configurations

The results of the preliminary analyses of the five basic configurations were evaluated in an effort to eliminate those designs which have very little potential for the Weight Lifter application.

It is evident that the effect of wings is almost totally detrimental for a transport vehicle concerned with relatively low speed and short range requirements which must also be capable of vertical flight and periods of hovering. The undesirable aspects of the wings stem primarily from their addition to the aircraft's gross weight, for which they make a comparatively small contribution to lift in the regime of flight speeds anticipated.

It may be assumed that interference effects will exist between the wing and ducted propellers. This interference would occur due to the influence that a highly loaded ducted propeller exerts on the airflow in a large vicinity about the duct. In turn, the presence of large lifting surfaces near a ducted propeller may be expected to produce varying effects on the ducted propeller inlet as the lift of the surfaces varies.

The use of pusher propellers as the source of propulsion for forward flight requires so much more horsepower than the tilting duct systems that the use of this method, with or without the presence of wings, appears to be unsatisfactory.

Both of the tilting duct no wing designs are superior to the three winged aircraft investigated. While the preliminary analysis indicated that the four-duct configuration would weigh somewhat more than the two duct design, they displayed many similar characteristics, and the merits of each were sufficient to warrant continued analysis of both configurations.

A more detailed weight breakdown, as shown in Section 5 of this report, was subsequently conducted for the no-wing configurations. This weight breakdown definitely established the four duct design as the heavier aircraft.

The curves of the fuel available to gross weight ratio, R_F , (shown in Figures 22 through 25) indicate that the two-duct no-wing design can carry the largest fuel supply for the disk loadings and payloads investigated. The larger fuel supply provides this design with the greatest range to gross weight ratio, which in turn is necessary to obtain the maximum value for the payload ton-mile per pound gross weight criterion discussed in Section 3.4.3.

The mutual interference between ducted propellers located in the vicinity of one another is a condition for which no method of evaluation is available. It is believed, however, that interference between the front and

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rear pairs of ducted propellers of the four-duct design is possible in forward flight. This interference would require additional power to replace any loss of effectiveness incurred.

The investigations conducted to determine an optimum Weight Lifter configuration indicate that the two ducted propeller, no-wing design is the most satisfactory design solution.

4.3 Further Design of Two-Duct Configuration

With the selection of the two-duct, no-wing configuration as the optimum Weight Lifter, further design studies were carried on for this configuration only. These studies were performed to optimize the individual functions of the aircraft with regard to performance, gross weight, utility, and simplicity.

4.3.1 Powerplant Location

For this study the only powerplant considered satisfactory to drive the ducted propellers was the geared gas turbine. This engine has a very high thrust to weight ratio and a rapidly improving specific fuel consumption. The specific fuel consumption assumed in this study was 0.55 lb/HP-hr, which has been predicted for an engine available in 1962 (References 2 and 5). No specific engine make was used; rather, average values from all engines in the horsepower range anticipated were determined and used for the evaluations.

Two locations were available for installing the engines of the two-duct no-wing configuration. The alternatives were to mount the engines in the centerbody of the ducts, or to install the engines in the fuselage.

The duct-mounted engines have the obvious advantage of direct drive through the gear boxes to the propellers, eliminating long drive shafts and angular drives with their complexity and heavy weights. Duct-mounted engines are located so that the engine weight counteracts a portion of the lift from the ducted propellers, thus reducing the duct supporting structure weight. The residual thrust from the engine is always directed so that it supplements the propeller thrust.

The duct-mounted engines also have certain disadvantages. The lengths of the shaft turbine engines applicable for this aircraft are in the 10 to 12 foot range, and the addition of a contrarotation gear box further lengthens the unit. Also, preliminary theoretical investigations being conducted by this contractor indicate that the best location of the propellers in a duct is very near the duct exit; thus it is difficult to install the engines in the duct centerbody and maintain clearance with the ground. A small ground clearance would not appear satisfactory because of the effects of the force and heat of the jet blast impinging directly on either prepared or unprepared surfaces.

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It is necessary to tilt the ducted propellers about a point on the duct in order to prevent the tilting of the duct from shifting the thrust vector to such an extent that an unsatisfactory moment is created by the misalignment of the thrust vector with the aircraft center of gravity, as shown in Section 6.7. The center of mass of the engines must then be raised through the arc of the duct tilt angle from its position below the ducts in hovering to a position aft of the ducts for forward flight. This center of mass located aft of the duct tilting pivot corresponds to a duct pitch-up moment, and thus adds to the aerodynamic duct pitch-up moment present in forward flight. Therefore, additional structural weight is necessary to withstand the moment imposed upon the duct tilting pivot when the ducted propellers are tilted for forward flight.

The ducted propeller efficiency would be lowered somewhat by the presence of the large centerbody containing the powerplants in the ducted propeller slipstream. This condition is similar to the effect of a nacelle placed behind an unshrouded propeller (References 11 and 12). The presence of a large centerbody in the duct reduces the effective duct area, and in order to provide the required mass flow through the duct, the duct diameter must then be increased to account for the loss in area due to the centerbody. Even a small increase in duct diameter requires a significant increase in gross weight due to the increase in duct circumference, propeller diameter, and gear box sizes.

The necessity for some degree of safety in the event of an engine failure dictates a multi-engine requirement, in order that some power be available to effect an emergency landing. Although two or more engines could be grouped together in the centerbody of each duct, the penalty for failure of a single engine would be equivalent to the loss of two engines, as power must be reduced on the opposite side of the aircraft to maintain a level attitude.

The location of the powerplants in the ducts nearly precludes any use of residual thrust or compressor bleed air for pitching moment reaction control purposes during hovering flight. A turbojet engine would be required to provide the necessary pitch control forces.

Fuselage-mounted powerplants have the advantage of a fixed position, with resultant lighter mounting weights and reduction of vibration problems. Installing the engines in the fuselage allows the duct centerbody to remain small, thereby reducing the duct centerbody losses and increasing the effective duct area.

Grouping the engines in the fuselage parallel to the longitudinal axis provides a situation where the residual jet flow of the several engines may be ducted together to the aft of the aircraft to provide additional thrust for forward flight and forces for reaction controls during hover-

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ing. By interconnecting the several engines necessary to provide the required horsepower, failure of one engine does not require a further reduction in power to maintain stability.

A serious problem confronting turbine engines operating from unprepared surfaces is the ingestion of foreign objects into the engine inlet. Installing the engines and the engine inlet duct at the top of the fuselage places the engine intake in the area most protected from objects thrown into the air by the ducted propeller slipstream.

The greatest difficulties encountered from use of the fuselage-mounted engines stem from the long drive shafts and extra gear boxes necessary to transmit the power to the propellers.

From the preceding analysis of powerplant location with regard to the Weight Lifter specifications, the fuselage-mounted engines were selected as the most satisfactory compromise of the factors involved. A preliminary weight analysis of the required drive shafting indicated a weight equal to approximately two percent of the aircraft gross weight (see Section 5). It may be assumed that the effect of this weight would be counteracted by the increased efficiency of the ducted propellers as a result of a small centerbody and the lighter ducted propeller assembly possible through the reduction in diameter.

5.3.2 Residual Thrust Utilization

The residual thrust from the shaft turbine powerplants was computed as an average of several makes of engines in the horsepower range contemplated, and was found to average 0.26 lb of thrust per shaft horsepower (References 3 and 13). As this thrust amounts to several thousand pounds, a study was undertaken to determine how this force might best be utilized to improve the aircraft's flight performance.

Assuming fuselage-mounted engines, the result of the study produced a system of ducting the residual jet flow of the several engines into a common tailpipe which extends to the aft of the aircraft. The tailpipe nozzle would act as a thrust diverter, and allow the thrust to act in a longitudinal direction for supplementing forward flight propulsion, or divert the thrust vector to produce control moments during hovering flight. This nozzle system may be visualized as a modification of the reaction control nozzle designed for the Hiller X-18 tiltwing aircraft (Reference 14). Incomplete tests of the X-18 deflection nozzle indicate ducting and turning losses of approximately seven percent. For this study, ten percent reaction control losses were assumed.

Sample calculations of the control moments required for hovering (see Section 6) indicate that sufficient residual thrust is available at the reaction control nozzle to provide pitch and yaw control as well as a balancing force to counteract the moment produced by the maximum allow-

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able thrust vector and aircraft center of gravity misalignment. Approximately one-third of the residual thrust can be utilized for the purpose of balancing any moment induced by ducted propeller thrust. By a judicious location of the aircraft center of gravity, the residual balancing thrust can be most often used as a lifting force during hovering. This hovering control system is considered quite efficient, as it utilizes a by-product (residual thrust) as its source of power, and offers multi-engine reliability as well. The alternate control systems often proposed for VTOL aircraft call for the addition of a turbojet engine or the use of turbine compressor air bleed, with their respective disadvantages.

Residual thrust acting parallel to the longitudinal axis in forward flight supplements the propulsive force of the ducted propeller; the resulting increase in forward velocity is quite significant, as may be seen by comparing Figures 6 and 15. As an additional advantage, the residual thrust produces a nose-down or stabilizing pitching moment (see Section 6.3.3).

6.3.3 Optimizing Disk Loading

The horsepower to gross weight ratio required for forward flight was recalculated, utilizing the effect of the shaft turbine residual thrust. The residual thrust was taken into account as a force acting opposite to the external drag force. The momentum equation used to determine the ducted propeller exit to inlet velocity ratio, ϵ , was modified to include the residual thrust, T_j , as

$$\epsilon^4 \left(1 - 0.5C_{D_i} \right)^2 - \epsilon^2 - \epsilon \left(C_{D_e} - \frac{T_j}{A_e V_o^2 \rho / 2} \right) = \tau^2 + 0.25 \left(C_{D_e} - \frac{T_j}{A_e V_o^2 \rho / 2} \right)^2$$

As $T_j = 0.26$ HP required, the above equation reduces to

$$0.903\epsilon^4 - \epsilon^2 - \epsilon \left(C_{D_e} - 0.0129 \sqrt{W_e} \right) = \tau^2 + 0.25 \left(C_{D_e} - 0.0129 \sqrt{W_e} \right)^2$$

Curves of ϵ and τ as a function of W are shown in Figures 13 and 14, respectively, from which the horsepower required for forward flight was calculated and plotted in Figure 15. Utilizing the residual thrust, the

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Increase in maximum velocity varied from 15 to 45 knots for disk loadings of 5 to 60 pounds per square foot, as can be seen in Figure 16.

The optimum disk loading was then determined for the Weight Lifter, considering the resultant thrust and its resultant increase in flight speed. The plot of gross weight as a function of disk loading and payload is shown in Figure 17, from which the optimum disk loading may be found at the minimum gross weight for each payload. It was found that the optimum disk loading remained at a constant value of

$$x_{opt} = 10.5 \text{ ft}^2$$

throughout the range of payloads investigated. Inasmuch as the horsepower required for this aircraft was determined from the hovering condition and the optimum disk loading applies to all gross weights, HP/W_0 is a constant. Therefore, the maximum velocity is the same for any size of the Weight Lifter configuration.

4.3.4 Propeller

The optimum disk loading of 10 pounds per square foot determines a range of propeller diameters from 12 to 6 feet for the range of gross weights involved, as shown in Figure 18. These diameters provide for the increased disk area required by the presence of a 0.7 propeller diameter center body. A Hiller survey of propeller sizes available by 1960 indicated that a maximum diameter of approximately 15 feet may be expected. It appears, then, that the range of propeller diameters specified for this study will be satisfied except for the highest gross weight machines. For this extreme of the gross weight range a slight increase in disk loading can be made, which would reduce the required propeller diameter to 15 feet with only a small increase in gross weight, as shown in Figure 19.

Because of their effects on the drive system and weight computations, counter-rotating propellers were assumed in the design study. Counter-rotating propellers are expected to provide an increase in propulsive efficiency of 10% above that of a single rotating propeller due to the absence of rotation of the propeller slipstream. It is also anticipated that the counter-rotating propellers will avoid large gyroscopic moments acting on the aircraft due to angular velocities of the aircraft in pitch or roll. Some degree of cross coupling between roll and yaw can be prevented by counter rotation, in that a differential thrust variation for roll will have no torque effects and thereby no resultant yawing moment.

4.3.5 Deeper Ducts

The use of counter-rotating propellers prompted a further investigation of the shroud length. The very short-chord shroud design originally designated for this study appears insufficient to properly enclose the

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large counter-rotating propellers necessary for the two-duct configuration. A further investigation of more recent studies (reference 15) indicates that longer chord ducts may be necessary to maintain high efficiencies. For this reason, 0.95 to 0.96 ducts are now specified for the Weight Lifter study.

3.6 Revised Drag Analysis

The use of longer ducts required a re-evaluation of the external drag. Because the drag analysis and results have been necessarily questionable, as discussed in Section 3.4.2, a further effort to establish a drag coefficient by the same method was not attempted; rather, the effect on forward flight performance was investigated for a range of external drag values. This study indicated that external drag has no effect on the value of optimum disk loading. Maximum forward flight velocity and range are the only characteristics varied by changing the external drag.

Because the optimum disk loading is independent of external drag, the hovering characteristics remain fixed. As the Weight Lifter study is based upon hovering performance, the resulting speed and range variation with external drag become secondary effects; therefore, the optimum configuration remains valid for the range of external drag values even though the exact maximum velocity and range is not known. The drag analysis indicated that a ten percent variation in external drag creates a corresponding seven percent variation in maximum velocity at the optimum disk loading.

As a result of the above considerations the forward flight performance remains based upon an external drag coefficient of

$$C_{D_e} = 0.24$$

3.7 Angle of Duct Tilt in Forward Flight

As mentioned previously, the optimum disk loading and maximum forward aircraft velocity are constant for the range of Weight Lifter gross weights. The value

$$\bar{V} = 0.98$$

is then a fixed value for the maximum velocity at all gross weights. The angle that the ducted propeller centerline is tilted forward from the vertical, α , can then be read from the plot of α vs $\bar{V}$, on the "residual thrust included" curve (Figure 18). The angle of forward tilt of the ducted propellers for maximum velocity is

$$\alpha = 55^\circ$$

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4.3.8 Mutual Interference Effects of Ducts in Close Proximity

An attempt to determine the magnitude of the mutual interference of ducted propellers operating in close proximity to one another was made in which the axial velocity component of a ducted propeller in the hovering condition was represented by the axial velocity component of a vortex ring (Reference 16). The axial velocity at a location corresponding to the centerline of one duct appears to be affected less than one percent as the result of the operation of the other ducted propeller. On the basis of this preliminary study, the effect of the mutual interference between ducted propellers in hovering was neglected. No method exists for determining the interference in forward flight.

4.3.9 Fuselage Design

The ability to load and transport bulky cargo is anticipated as forming the major part of a Weight Lifter's operation, and this requirement dictates a large-volume cargo compartment for which a minimum amount of special loading equipment would be necessary. A preliminary survey of existing cargo aircraft was made, and it was determined that for the Weight Lifter, the cargo compartment volume to payload weight ratio would be approximately two to three times that of present military fixed wing cargo aircraft (Reference 17), and somewhat greater than helicopter cargo aircraft (Reference 18).

The fuselage design was modified so as to make the best use of the fuselage engine mounting and tailpipe system; the final design, then, became a large capacity pod-type body with a single large boom containing the engines and tailpipe extending aft from the top of the pod. The empennage is mounted above the aft end of the boom, to remain as free as possible from the effects of the ducted propeller wake.

Cargo may be loaded from both the front and rear of the fuselage, with an oversized door and loading ramp at the rear. For the larger aircraft designed for the higher payloads, the fuselage height is sufficient to allow a hinged floor to be positioned so as to provide two levels for transporting large numbers of troops.

4.3.10 Cockpit Design

It is foreseen that the Weight Lifter would perform many operations similar to those required of large helicopters; therefore, it is necessary to provide pilot visibility comparable to that of a helicopter. The cockpit was extended forward beyond the main fuselage structure, providing vision downward and to the sides through a helicopter-type transparent nose section.

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4.3.11 Duct Support System

The ducted propeller supporting structure was redesigned to avoid a single pivot point mounting on the side of the duct as shown in Figure 18. With the heavy thrust loads taken out of the duct at only one point on its circumference, this support system would require a heavy duct structure to maintain duct rigidity. The new support structure became a V-strut from the top of the fuselage to the duct centerbody where it joined a horizontal member from the center of the fuselage, as shown in Figure 19. For this support arrangement, the duct tilts about the horizontal member while pivoting on the V struts at the duct centerbody. The support structure was placed above the duct in an effort to minimize the drag effects by locating the struts in the lower velocity airstream about the inlet rather than in the higher velocities encountered at the duct exit. A definite weight saving was realized as a result of this redesign.

4.3.12 Allowable CG Variation

As a result of studies of the thrust vector and the residual thrust balancing potential, it was determined that the aircraft center of gravity could be allowed to vary two feet longitudinally while maintaining a controllable moment at both hovering and forward flight conditions. This CG shift is comparable to that limit established for present fixed wing cargo aircraft. A vertical shift in CG appears not to be critical in its relationship with the thrust vector throughout the required range of duct tilt angles.

4.3.13 Propeller Wake Deflection

As mentioned previously, a conventional horizontal stabilizer and elevator are required for the optimum Weight Lifter configuration to provide longitudinal stability and pitching moment control during forward flight. The effectiveness of the horizontal tail is dependent on the ducted propeller wake conditions at the tail location.

In order that the tail section might be placed out of the direct downwash effect of the slipstream the slipstream deflection of the ducted propeller in forward flight was investigated. The deflection was only approximated, due to the lack of information concerning the characteristics of the airflow in the vicinity of a highly loaded ducted propeller.

The slipstream was approximated as the resultant of the ducted propeller exit velocity and aircraft velocity components, as shown in the velocity diagram in Figure 21. The angle δ , between the horizontal and the resultant slipstream can then be determined as a function of Ψ (Figure 21). The most serious condition for providing clearance between the tail and the slipstream occurs at the maximum forward flight velocity. At this

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condition, the angle δ is a minimum, and the rear of the duct is raised to its highest position. In this situation, with the slipstream deflected immediately upon leaving the exit at the rear of the duct, it was determined that the flow still passed well beneath the horizontal tail location.

4.4 Further Design of Four-Duct Configuration

For purposes of comparison, those design features discussed in Section 4.3 which were applicable to a four-duct configuration were incorporated in that design, as shown in the drawing in Figure 20. On the basis of these changes, a revised weight analysis for the four-duct configuration was made, as discussed in Section 4.2.

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5. WEIGHT ANALYSIS

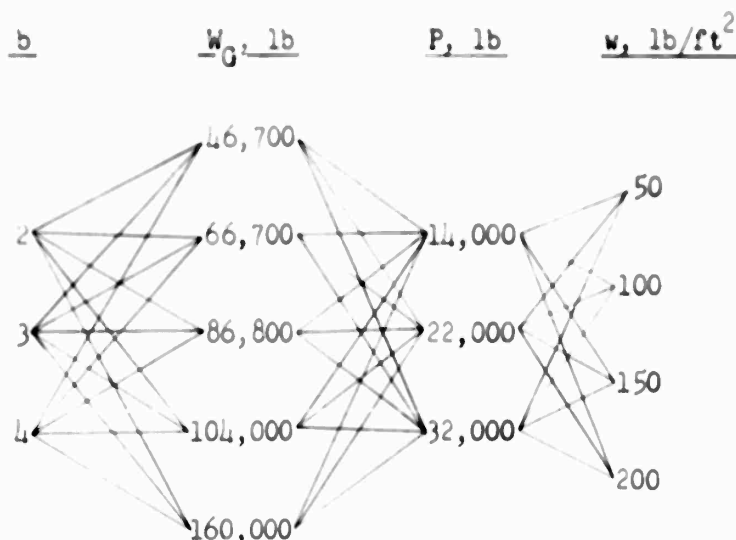
5.1 Introduction

The objective of this analysis is to compare the five Weight Lifter configurations on the basis of aircraft gross weight. Because of the lack of certain aerodynamic as well as structural design information on this type of aircraft at present, a minimum number of design parameters were chosen to be used in the R_F Graphical Method (Reference 10) for developing an optimum design.

5.2 Parametric Study and Application of R_F Available

A selection of four parameters is made in this study for the optimization of a preliminary Weight Lifter design. These four parameters, as shown in matrix form below, are numbers of ducts, b ; gross weight, W_0 ; payload, P ; and disk loading, w .

PARAMETRIC STUDY



In the " R_F " Graphical Method, R_F , ratio of fuel weight to gross weight, is used as a solution for a pair of simultaneous equations, namely, the " R_F required" of the aerodynamic analysis and, " R_F available" of this section. The point of intersection in each case represents the least gross weight for that combination of payload, disk loading, and configuration. The optimization of a series of such combinations yields the lowest gross weight, as shown in Figures 10 and 11.

As always, the cost of an aircraft is directly associated with its weight; therefore, the advantage of choosing a configuration in preliminary design from the weight standpoint is obvious. This method also eliminates

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much laborious work involved in optimizing all the possible combinations. To facilitate selection of the best configuration, this "Rp" Graphical Method is further developed in Figures 22 through 25 by comparing the minimum gross weight curves for different disk loadings and payloads. The final outcome is that the two duct no wing configuration appears as the best. The result of the optimization study by the Rp method is listed below.

<u>Payload</u>	<u>Two Ducts</u>	<u>Four Ducts</u>
15,000 lb	$W_0 = 16,700 \text{ lb}$ $w = 11 \text{ lb/ft}^2$	$W_0 = 61,000 \text{ lb}$ $w = 75 \text{ lb/ft}^2$
22,000 lb	$W_0 = 29,000 \text{ lb}$ $w = 50 \text{ lb/ft}^2$	$W_0 = 108,000 \text{ lb}$ $w = 60.3 \text{ lb/ft}^2$
32,000 lb	$W_0 = 11,500 \text{ lb}$ $w = 60 \text{ lb/ft}^2$	No Data

5.3 Weight Equations

For the conventional components and equipment of the aircraft, suitable weight equations were adopted from Reference 10. If existing equations did not apply, an adjustment was made by actually designing the unconventional component. A new weight equation was then introduced, either by modifying the equations found in references to suit the provisional design or by developing an entirely new equation. However, for the sake of accuracy the new equations generally were tested against those for configurations of similar weight.

In primary structures design an ultimate vertical load factor of 3.0 is applied individually to the subjected component, while a vertical load factor of 2.0 is considered plus either a side load factor of 1.5 or a fore-aft load factor of 1.5. The allowables for stress are per Reference 18. For this preliminary design, thermal stress is not investigated, as its effect will be negligible for the skin friction generated at the anticipated flight speeds. Also, aeroelastic problems are not included, as they are beyond the scope of this study.

In deriving the Rp available equations, all the constants possible are predetermined by a survey of other similar aircraft.

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Crew Weight = 600 lb

Tip Speed of Propeller = 800 ft/sec

All the operation is assumed to be performed at sea level under standard conditions:

$$\rho = 0.002378 \text{ slug/ft}^3$$

The equation for the gross weight of the aircraft can be written as

$$W_G = W_{\text{empty}} + W_C + P + W_F + W_{FT}$$

where

W_{empty} = empty weight of aircraft, lb

W_C = crew weight, lb

P = payload, lb

W_F = fuel weight, lb

W_{FT} = fuel tank weight, lb

The weights of components comprising total empty weight of the aircraft are listed as follows:

W_R = propeller weight (rotor weight), lb

W_D = duct weight, lb

W_W = wing weight, lb

W_{EM} = empennage weight, lb

W_B = fuselage weight, lb

W_L = landing gear weight, lb

W_P = powerplant and accessories weight, lb

W_{FE} = fixed equipment weight, lb

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W_M = miscellaneous items weight*, lb

The ratio of aircraft empty weight to gross weight is expressed by

$$\phi = \frac{\sum \text{Component Weights}}{W_G}$$

The weight of fuel and fuel tank is obtained by

$$W_F + W_{FT} = W_G - P - W_{\text{empty}} - W_C$$

Since W_{FT} is a function of W_F , then

$$W_F = K(W_F + W_{FT})$$

The assumption is made that the jet fuel weight is 6.5 lb per gallon, and the tank weight is 0.5 lb per gallon; therefore

$$K = 0.928$$

Hence

$$\frac{W_F}{W_G} = 0.928 \left(1 - \frac{P}{W_G} - \frac{W_{\text{empty}}}{W_G} - \frac{W_C}{W_G} \right)$$

or

$$R_F = 0.928 (1 - R_P - \phi - R_C)$$

5.3.1 Rotor Weight, W_R

$$W_R = 0.757 \frac{W_G}{w} \left(\frac{AF}{100} \right)^{1.8} (B_1)^{.825} \quad \text{See Reference 5} \quad (1)$$

Where

AF = activity factor

W_G = gross weight, lb

* This term includes tilting mechanism, tail boom, and the miscellaneous items shown in Sections 5.3.15 through 18.

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w = disk loading by rotor area = $\frac{T}{A}$, lb/ft²

A = area swept by rotor, ft²

B_1 = number of blades, expressed as

$$B_1 = \frac{(1,360)(1 \cdot f)w_e}{(AF)C_{L_R} V_T^2} \left(\frac{A_1}{A_2} \right)^2 \left[\frac{2.59}{\sqrt{1 \cdot \left(\frac{V_2}{V_T} \right)^2}} + \frac{1.0}{\sqrt{0.09 \cdot \left(\frac{V_2}{V_T} \right)^2}} \right] \quad (2)$$

See Reference 5

Where

$1 \cdot f$ = flow area = equivalent flat plate area of drag surfaces = 1.25 ft²

C_{L_R} = mean blade lift coefficient = 0.53 for optimum C_L/C_D

$\frac{A_1}{A_2}$ = the ratio of slipstream area downstream to that of the propeller = 1.0 for ducted propellers with straight exit ducting

V_2 = downwash velocity, ft/sec = $\sqrt{\frac{w_e}{\phi}}$

w_e = effective disk loading, lb/ft²

V_T = tip speed = 800 ft/sec

An AF value of 100 was chosen. This agrees with conventional reciprocating engine propellers with disk loadings of about 85 lb/ft², which corresponds to the average disk loading in this study.

Then from equation (1)

$$w_R = 0.0314 \frac{w_G}{w} \left[w_e \left(\frac{10,200}{1520 + w_e} \right)^{\frac{1}{2}} + \left(\frac{1,520}{137 + w_e} \right)^{\frac{1}{2}} \right]^{.825} \quad (3)$$

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since

$$\lambda_e = \frac{W_G}{W_e} \text{ for flow area of single duct}$$

and

$$\frac{\lambda_e}{\lambda} = \frac{\text{flow area}}{\text{rotor area}} = \frac{1 - (.3)^2}{1.0} = 0.91$$

For hovering

$$W_e = \frac{T}{\lambda_e} = \frac{T}{0.91\lambda} = 1.1W$$

The disk loading per hub, $\frac{W_e}{H}$ is used for a counter-rotating propeller

where

$$H = 2 \text{ for counter-rotation}$$

Thus (3) becomes

$$W_R = 0.0375 \frac{W_G}{(W) \cdot 175} \left[\left(\frac{10,200}{1,520 + .55W} \right)^{\frac{1}{2}} \cdot \left(\frac{1,520}{137 + .55W} \right)^{\frac{1}{2}} \right]^{.825} \text{ per ship.}$$

NOTE: W_R is independent of b , number of ducts, which must not equal zero in any case.

5.3.2 Transmission Weight, W_T

The transmission weight is based on the hovering condition. Reference 2 gives

$$W_T = (0.081Q)^{.88} \left(\frac{n+1}{2} \right)^{.375} \quad (4)$$

where

$$Q = \text{output shaft torque, ft-lb} = \frac{5250HP}{N}$$

n = number of installed engines

$$N = \text{rpm} = \frac{60V_t}{\pi D} = \frac{(60)(800)}{\pi D} = \frac{13,620}{(A) \cdot .5}$$

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D = diameter of propeller, ft

For M = figure of merit for static thrust = 1.28

$$HP = \frac{W_G}{550M} \sqrt{\frac{w_e}{2\rho}} = \frac{W_G(w_e)^{.5}}{40.76 b} \quad \text{per duct}$$

Therefore

$$Q = 0.00833 \left(\frac{W_G}{b} \right)^{1.5}$$

Substituting Q in (4)

$$\begin{aligned} W_T &= 0.0012 \left(\frac{n+1}{2} \right)^{.375} \left(\frac{W_G}{b} \right)^{1.32} \quad \text{per duct} \\ &= 0.0012 \left(\frac{n+1}{2} \right)^{.375} \frac{(W_G)^{1.32}}{(b)^{.32}} \quad \text{per ship} \end{aligned}$$

5.3.3 Duct Weight, W_D

Maintaining maximum stiffness at the propeller station of the duct is the major criteria for the duct weight estimation, as any rubbing of propeller blade tips against the shroud inner surface must not be permitted. Therefore, the weight of duct varies with different types of supports.

- a. For ducts built into the fuselage, as in the three-duct configuration, only the weight of centerbody struts, (about 35% of total duct weight) remains the same as that for the strut-mounted ducts discussed in "b" below. Total duct weight is reduced, as it partially becomes an integral part of the fuselage.

$$W_D = 0.0665(A)^{.51} (W_G)^{.74}$$

- b. For two or four duct configurations the duct support adds little stiffness.

Reference 1 expresses an equation for a similar configuration.

$$\begin{aligned} W_D &= 0.0542(D)^{2.6} (w_e)^{.7} \\ &= 6400 \text{ lb} \end{aligned}$$

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where $D = 38.36$ ft and

$$w = 30 \text{ lb/ft}^2$$

A check is made with Reference 3 for a conservative estimation.

$$L_D = 38.36 \text{ ft}$$

$$U_D = 1.77 (L_D) = 68.1 \text{ ft}$$

$$B_D = 5.18 \text{ ft}$$

$$H_D = (0.007)(38.36) = 0.27 \text{ ft}$$

$$i_D = \frac{(5.18 + 0.27)}{1.77} = 3.0 \text{ ft}$$

Then

$$i_D(B_D + H_D) = (3.0)(5.18 + 0.27) = 16.8 \text{ ft}$$

By Figure 3d of Reference 1

$$W_D = 6.00 \text{ lb for } V < 300 \text{ knots}$$

Comparing with results from Reference 1

$$\frac{6.00}{6.00} = 1.0$$

The weight of 6.00 lb from Reference 1 seems to be too conservative and as a compromise, exponents were readjusted.

$$W_D = 0.0512(D)^{1.5}(w)^{0.74} \text{ is adopted.}$$

$$W_D = 0.0512 \left(1.36(A)^{1.25} \right) \left(\frac{W_G}{bA} \right)^{0.74}$$

$$= 0.074(A)^{0.51} \left(\frac{W_G}{b} \right)^{0.74} \text{ per duct}$$

$$= 0.074(A)^{0.51}(b)^{0.26} (W_G)^{0.74} \text{ per ship} \quad (5)$$

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5.3.4 Fuselage Weight, W_B

The fuselage weight equation for a helicopter is expressed as

$$W_B = 0.159 (W_G L)^{.69} \quad (\text{Reference 2}) \quad (6)$$

For conventional fixed-wing fuselage design, the weight equation is

$$W_B = 0.159 (W_G)^{.69} [L(H \cdot B)]^{.375}$$

where H = height, B = width, and L = length of fuselage

A sample calculation is shown below. To insure a conservative weight estimate and to provide maximum cargo space, these values were chosen:

Let $H = 3 = 0.16 L$

For $W_G = 100,000$ lb and $L = 100$ ft

$W_B = 9,200$ lb, which agrees with the weight estimation in Figure 38, Reference 3

NOTE: For the four-duct configuration, because of structural requirements

$$W_B = 0.159 (W_G)^{.69} [L(B \cdot H)]^{.39}$$

5.3.5 Wing Weight, W_W

From Reference 3, utilizing a conventional type wing, the expression for wing weight becomes

$$W_W = \frac{2.31 b_1}{f_1} W_G (4.95 + 3.6) = \frac{19.75}{f_1} W_G b_1 \quad (7a)$$

where

b_1 = span, ft

f_1 = design stress factor, see Figure 35, Reference 3

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which, for the example W_G of 100,000 lb and a low wing loading of 50 lb/ft² reduces to

$$W_W = \frac{6.92}{F_1} (W_G)^{1.5} \quad (7b)$$

5.3.6 Horizontal Tail Weight, W_{HT}

The weight analysis for the horizontal tail is the same as for the wing. Equation (7a) then reduces to

$$W_{HT} = \frac{2.78}{F_1} (W_G)^{1.5} \quad (\text{two ducts on wingtip})$$

$$W_{HT} = \frac{2.50}{F_1} (W_G)^{1.5} \quad (\text{two ducts on side})$$

5.3.7 Vertical Tail Weight, W_{VT}

Since vertical tail weight is a function of vertical tail area, speed, and atmospheric conditions,

$$W_{VT} = K A_t^m q^n$$

Then, based on Figure 37 of Reference 3

$$W_{VT} = 0.064 (A_t)^{1.15} (V_a)^{0.5} (\sigma)^{0.25} \quad (8)$$

where $\sigma = \frac{\rho}{\rho_0} = 1.0$ at sea level

V_a = velocity of aircraft, knots

A_t = vertical tail area, ft²

The equation can be further reduced by the substitution

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$$A_L = KL^2$$

where

L = length of fuselage

For design purposes, $K = 0.027$ for the two ducts on wingtip type, and $K = 0.033$ for the two-duct no-wing type, by aerodynamic requirement.

5.3.8 Landing Gear Weight, W_L

For fixed landing gear the weight was determined by the relation

$$W_L = 0.0337 (W_G)^{1.02} \quad (\text{Reference 2})$$

5.3.9 Engine Weight, W_E

Chart I of Reference 1 predicts the weight of shaft turbine engines up to 1965. From this data a value of 0.32 lb/HP is assumed for engines produced later than 1962.

$$W_E = 0.32 \text{ HP}$$

$$= 0.32 \left(\frac{1.2 W_G (w_e)^{.5}}{46.2} \right) = 0.0089 W_G (w)^{.5}$$

5.3.10 Engine Section Weight, W_{ES}

$$W_{ES} = 0.053 \left(\frac{W_G}{LP_m} \right)^{1.07} (n)^{-.07} \quad (\text{Reference 2})$$

$$LP_m = \frac{W_G}{1.167 \text{ HP}} = \frac{W_G}{0.0317 W_G (w)^{.50}} = \frac{1}{0.0317 (w)^{.50}}$$

$$W_{ES} = 0.00126 (W_G)^{1.07} (w)^{.535} (n)^{-.07}$$

where n = number of engines $\neq 0$

5.3.11 Starting System Weight, W_{SS}

$$W_{SS} = 0.29(n)^{.4} \left(\frac{W_G}{LF} \right)^{.6} \quad (\text{Reference 2})$$

$$= 0.035(W_G)^{.6} (w)^{.3} (n)^{.4}$$

5.3.12 Oil and Oil Tank Weight, W_O

$$W_O = 0.04 W_E \quad (\text{Reference 2})$$

$$= 0.00036 W_G (w)^{.5}$$

5.3.13 Fixed Equipment Weight, W_{FE}

This term includes weight of instruments, flight controls, hydraulic and electrical systems, furnishings, and communications equipment.

$$W_{FE} = 1.93(W_G)^{.672} \quad (\text{Reference 2}) \quad (9a)$$

However, in order to include the weight of a helicopter type cabin, Equation (9a) is modified to

$$W_{FE} = 2.38(W_G)^{.672} \quad (9b)$$

5.3.14 Boom and Exhaust Pipe Weight, W_{BP}

This component is treated as a cantilever beam loaded by its own weight, tail load, and thrust from the reaction controls. It is assumed that the longerons are supported at intervals by light frames to insure stability. For aerodynamic reasons a light sheet is chosen for the housing skin, as a smooth surface is of greater importance than structural stability. Because of the low speed range of these designs, the thermal stress due to boundary friction heating is not assumed to damage the skin. For practical design, the exhaust pipe and its insulation are estimated to weigh 19.3 lb/ft, and the boom to weigh 3.7 lb/ft (based on a 100,000 lb class aircraft).

Then

$$W_{3P} = 0.021 W_G$$

5.3.15 Gear Box Weight, W_E

The weight of the primary reduction gear box is included in the transmission and engine section, but the weight of the intermediate gear box is treated separately and estimated to be

$$W_E = 0.13 \text{ lb/HP}$$

5.3.16 Shaft Weights, W_S

The main transmission shaft is designed to withstand both the torque and its own weight. It is also checked against its frequency to avoid any resonance which would lead to vibration. From these considerations

$$W_S = 4.25 \text{ lb/ft}$$

5.3.17 Tilting Mechanism and its Accessories Weight, W_{TM}

$$W_{TM} = 0.0045 W_G \text{ per duct for two-duct configuration}$$

$$= 0.0030 W_G \text{ per duct for four-duct configuration}$$

The constant is a result of a statistical survey, and is recommended due to the fact that the screw jack is not only operated by a hydraulic system but also by an electric motor, with manual control in case the hydraulic system fails. Therefore, this estimated weight is justified for the importance of its function.

5.3.18 Duct Supporting Strut Weight, W_{DS}

All the compression members in the strut system are pipes of steel alloy with $D/t < 50$ to avoid a local crippling problem. This weight then becomes

$$W_{DS} = 0.015 W_G$$

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5.4. Summary

5.4.1 R_p Available Graphs

The R_p available curves in Figures 22 through 25 show that R_p available is a non-linear function of either disk loading or gross weight. Empty weight is a function of gross weight, and numerical calculations show that a variation in R_p available changes only the value of the empty weight to gross weight ratio, $\emptyset$, as may be seen from the equation

$$R_p = 0.925 (1 - R_p \quad R_G - \emptyset)$$

in which crew weight and payload can be held constant. The relationship between R_p available and disk loading is therefore also non-linear, as disk loading is a function of empty weight.

The non-linearity of these relationships is a result of the complexity of the weight equations for the empty weight components, and any increase in either disk loading or gross weight or both does not produce a corresponding increase in R_p available. However, the ratio of R_p available to gross weight for the disk loadings ranging from 50 to 100 lb/ft² is nearly linear up to a gross weight of 100 000 pounds. The R_p available equations for the empty weight components of the final configuration will be found in Appendix 5 A.

Examinations of the R_p plots in Figures 22 through 25 reveal optimum points at disk loadings which produce rather low forward speeds and short ranges. As explained in Section 3.4, an operations research investigation produced an optimization based on a criterion of maximum payload ton-miles per pound gross weight, also determined from these R_p curves. This investigation indicated that an aircraft capable of higher speeds and longer ranges would be more efficient for the anticipated Weight Lifter mission, and the design was adjusted accordingly.

5.4.2 Selection of Configuration by Weight Analysis Based on Same HP, w , P and W_G

Two-Duct on Wingtip Configuration Figure 1 In general, the two-duct configurations have a better R_p and R_F than the three-duct delta wing, as the cargo space is better utilized in the fuselage. It is structurally a transport, conventional both in arrangement and construction except for the rotating ducts on the wing tips. Because of the redundancy in providing the lift necessary for forward flight by both duct and wing, the combined weight does not recommend this configuration.

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Two-Duct Canard Configuration, Figure 2: This design is eliminated for the same reasons as the configuration shown in Figure 1.

Three-Duct Delta Wing, Figure 3: This configuration is regarded as a poor prospect, as its M_p and M_f are too low to be considered practical. The main reason for the small useful load is the fact that structurally it demands a redundancy of members to strengthen the three cut-outs for installing ducted propellers in the delta wing. In addition, the triangular arrangement of the three ducts leaves no optimum available space for internal cargo loading.

Four-Duct No-Wing Configuration, Figure 5: This type was studied because of its simplicity, being without either wing or horizontal tail. However, the excessive weight of its aft fuselage section subjects it to the same criticism as the delta wing, and the extra set of ducts and their accessories cancel out its advantage over the two-duct configurations.

Two-Duct No-Wing Configuration, Figure 4: This type features two ducts on struts, one on each side of the fuselage. The elimination of the entire wing structure saves considerably in weight, and the use of a tail boom instead of aft fuselage section is also weight-saving. Therefore, this design is chosen as best on a weight basis.

For purposes of comparison, a tabular weight component summary will be found in Appendix 5-B for the configurations shown in Figures 1, 4, and 5.

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APPENDIX 5-A: R_p AVAILABLE EQUATIONS

Two-Duct No-Wing Configuration

$$R_p = 0.928 (1 - R_C - R_p - D)$$

$$R_C = \frac{600}{W_G}, R_p = \frac{P}{W_G} \text{ and } D = R_1 + R_2 + \dots + R_{12}$$

where

$$R_1 = \frac{0.0375}{(w)^{.175}} \left[\left(\frac{10,200}{1,520 \cdot 0.55w} \right)^{.5} + \left(\frac{1,520}{137 \cdot 0.55w} \right)^{.5} \right]^{.325}$$

$$R_2 = 0.0164 \frac{(L)^{2.3}}{W_G}$$

$$R_3 = 0.159 \left((B \cdot H) L \right)^{.375} (W_G)^{-.31}$$

The assumption is made that $H = B = 0.265L$, as the length of the fuselage in Figure 4 is assumed to be 60% of that of the other configurations, because of its tail boom.

$$R_4 = 2.38 (W_G)^{-.323}$$

$$R_5 = 0.089 (A_d)^{.51} (W_G)^{-.26}$$

$$R_6 = 0.0354 \left(\frac{n}{W_G} \right)^{.41} (w)^{.3}$$

$$R_7 = 0.034 (W_G)^{.02}$$

$$R_8 = 0.0092 (w)^{.5}$$

$$R_9 = 0.00126 \left(\frac{W_G}{n} \right)^{.07} (w)^{.535}$$

$$R_{10} = 0.00096 (W_G)^{.32}$$

$$R_{11} = \frac{2.5}{f_1} (W_G)^{.5}$$

$$R_{12} = 0.048$$

Components

Rotor

Fin

Fuselage

Fixed Equipment

Duct Assembly

Starting System

Landing Gear

Engine, oil and oil tank

Engine Section

Transmission

Horizontal Tail

Miscellaneous Items

APPENDIX 5-B: WEIGHT COMPONENTS SUMMARY
(Gross Weight = 100,000 lb, Disk Loading = 100 lb/ft²)

Component	Two-Duct No-Wing Configuration		Two Ducts on Wingtip Configuration		Four-Duct No-Wing Configuration	
	lb	lb	lb	lb	lb	lb
1) Fuselage		7,500		9,130		9,500
2) Wing				6,520		
3) Empennage		2,460		2,770		
4) Duct Group						
a. Duct Assembly	11,720		11,720		16,300	
b. External support	1,500				1,530	
c. Tilting mechanism	900	14,120	1,200	12,920	1,200	19,030
5) Powerplant Group						
a. Engine*	9,200		9,200		9,200	
b. Transmission	3,820		3,820		3,660	
c. Starting system	350	13,370	350	13,370	450	13,310
6) Engine Section		2,890		2,890		2,110
7) Rotor Assembly		6,690		6,690		6,690
8) Landing Gear		4,280		4,280		4,280
9) Tail Boom and Reaction Control		2,400				2,400
10) Fixed Equipment**		5,330		5,330		5,330
11) Exhaust Pipes				600		
12) Misc. (longshaft, etc.)						200
TOTAL EMPTY WEIGHT		59,040		64,500		63,450
13) Useful Load						
a. Fuel and fuel tank		18,360		12,900		13,950
b. Payload		22,000		22,000		22,000
14) Crew (three)		600		600		600
GROSS WEIGHT		100,000		100,000		100,000

* includes engine, oil and oil tank, cooling, lubricating and fuel system.

** includes instruments, flight control, hydraulic and electrical system, furnishings, communication equipment.

6. STABILITY AND CONTROL

The stability and control characteristics of the optimum Weight Lifter configuration have been determined for a representative aircraft which carries the average payload of 22,000 pounds, as shown in Figure 19. As the aircraft has been designed to grow uniformly larger with increasing payload, the stability and control functions calculated for the average size aircraft are, in most cases, directly applicable to the other sizes. Those characteristics peculiar to aircraft at the extremes of the size range were checked to insure the suitability of the principle involved for a Weight Lifter throughout the range of gross weight.

6.1 Pitching Moment of the Fuselage

The pitching moment of the fuselage was calculated by Munk's method, corrected for body fineness ratio (Reference 12). The accuracy of this result was considered satisfactory, for the effect of the ducted propeller airflow on the fuselage moments is not known. As the fuselage remains at zero angle of attack for all equilibrium conditions, the criterion established for maximum fuselage pitching moment was the effect produced by a 50 foot per second vertical gust encountered at the aircraft maximum forward velocity of 150 knots. This condition results in an angle of attack of approximately 11 degrees. The maximum fuselage pitching moment for the representative Weight Lifter is then

$$M_F = 235,000 \text{ ft-lb}$$

The gust (and, therefore, the resulting angle of attack) was taken so as to produce a positive pitching moment in order to add to the other unstable moments. In this manner the maximum stability requirement of the aircraft is depicted.

6.2 Pitching Moment of the Duct

The pitching moment of the ducted propellers was determined from an investigation of the available test data. The Hiller Helicopters Flying Platform truck test pitching moment data (Reference 4) presented in Figure 26 represents the forward flight pitching moment characteristics of a hovering or static type shroud. The NACA test data (Reference 22), also presented in Figure 26, represents the pitching moment of a high speed type shroud. As previously mentioned, the shroud envisioned for the Weight Lifter would compromise those extremes; therefore, an average value of pitching moment coefficient was approximated for the Weight Lifter ducted propeller, as shown in Figure 26.

References 4 and 22 show a significant decrease in pitching moment with increasing ducted propeller forward tilt angle. This further substantiates

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the decision that tilting the ducted propeller is the most satisfactory solution for producing a forward propulsive force.

From the assumed Weight Lifter duct pitching moment curve, Figure 26 the resulting pitching moment, M_p , of the ducted propellers at the maximum aircraft velocity and the gust condition described in Section 6.1 becomes

$$M_p = 377,000 \text{ ft-lb}$$

as a nose-up or positive moment.

6.3 Pitching Moment due to Thrust Vector

The pitching moment of the aircraft due to the displacement of the thrust vector from the aircraft center of gravity can be found from the thrust vector and the allowable CG travel defined in Section 4.3.12. As it is more efficient to provide longitudinal equilibrium in hovering or vertical flight with a lifting force at the tail section, the largest portion of the CG range was established aft of the vertical thrust vector. This arrangement causes a nose-up pitching moment which requires a lifting force from the residual thrust nozzle for equilibrium. The CG travel was limited to 1.5 feet aft of the centerline of the ducted propellers in the hovering position and 0.5 feet forward of the centerline. This proportioning requires only small downward forces from the reaction controls at the most forward CG, and reasonably obtained upward reaction forces at the most aft CG location. It also utilizes a lifting force at the tail throughout the greatest portion of the CG range.

This range of CG locations was found to be quite feasible. The three-view drawing of the representative Weight Lifter shown in Figure 19 depicts the general arrangement of the major components which provides the proper center of gravity range.

The pitching moment due to the misalignment of the thrust vector and CG, M_T , for the maximum forward velocity becomes

$$M_T = 175,000 \text{ ft lb}$$

for the most aft CG location.

6.4 Pitching Moment from Residual Thrust

The residual thrust is utilized both in hovering and forward flight as a stabilizing moment. During hovering and vertical flight, the pitching moment produced by the reaction control system must counteract the moment created by the misalignment of the thrust vector and the aircraft CG (Section 6.3). The representative Weight Lifter requires 2,600 pounds

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reaction control force to balance this thrust produced moment at the most aft CG location. During forward flight the residual thrust vector passes above the aircraft CG thereby contributing to the aircraft's stability. The moment produced by the residual thrust is found to be

$$M_R = 74,000 \text{ ft-lb}$$

at the maximum flight velocity.

6.5 Pitching Moment of the Horizontal Tail

Under conditions of maximum forward velocity and a 50 ft/sec vertical gust, the total pitching moment contribution from the ducted propellers, thrust misalignment, and fuselage becomes 76,000 ft-lb as a nose-up or positive moment. A negative pitching moment due to the residual thrust reduces the forward flight positive pitching moment to 108,000 ft-lb, which must be balanced by the horizontal tail in order to establish equilibrium in pitch.

The pitching moment of the horizontal tail was based on a lift curve slope of 0.06 for an aspect ratio 1.5 tail surface (Reference 14). It was assumed that the tail had an efficiency of

$$\eta_t = 0.9$$

This efficiency is quite arbitrary due to the lack of information concerning the wake of a highly loaded ducted propeller. It is possible that a variable incidence horizontal tail would be required to counteract the variation in downwash as the ducted propeller tilts to produce various flight speeds. The horizontal tail moment arm for the representative aircraft is 55 feet to the most rearward CG location. From the above values, a horizontal tail area was determined which provides the necessary equilibrium.

The preceding pitching moment characteristics of the Weight Lifter require a horizontal stabilizer of approximately 350 square feet to produce an equilibrium condition throughout the range of flight speeds. Due to the gross assumptions required to estimate the moments of a ducted propeller type aircraft, no attempt was made to design the horizontal stabilizer to meet static and dynamic stability requirements; rather, it was felt sufficient to determine that a reasonably sized horizontal stabilizer could apparently overcome the unstable pitching moments generated by the ducted propellers and fuselage.

6.7 Rolling Moment

The rolling moment in forward flight is generated by differential thrust of the ducted propellers. This differential thrust is accomplished by changing the collective pitch of the ducted propellers.

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6.8 Yawing Moment

When roll control is applied during forward flight, the roll-yaw cross coupling will produce a yawing moment of less than 0.2 of the applied rolling moment. This results from the fact that while the ducted propeller is tilted forward as much as 55 degrees, the resultant thrust vector is tilted forward less than 10 degrees. The yawing moment induced by the roll control can be counteracted easily by a 1,00 pound force from the vertical tail.

6.9 Forces Necessary for Control During Hovering and Vertical Flight

An investigation of the control gradients available during hovering for helicopters and other proposed vertical take-off cargo transports was conducted (References 3, 4, and 5) to determine the forces necessary for adequate control of the Weight Lifter during hovering and vertical flight. The accelerations determined as a result of this preliminary study are the following:

- | | |
|--------------------------------------|--------------------------------|
| (1) Pitch reaction control | 0.3 radian/second ² |
| (2) Yaw reaction control | 0.1 radian/second ² |
| (3) Roll differential thrust control | 0.2 radian/second ² |

The pitch reaction control force required to produce a gradient of 0.3 radian/second² is 3,900 pounds acting at the residual thrust nozzle at a distance of 69 feet from the most aft aircraft CG.

The yaw reaction control force required to produce a 0.1 radian/second² gradient is 3,400 pounds acting at the residual thrust nozzle.

As mentioned previously 2,600 pounds of force at the reaction control nozzle is required to balance the ducted propeller thrust misalignment with the aircraft CG.

The total force required from the reaction control system as a result of applying the pitch, yaw, and thrust balancing forces concurrently was determined to be 6,600 pounds. With the inclusion of the 10 percent ducting and nozzle turning loss (Section 4.3.2) the residual thrust required from the turboshaft powerplants must be a minimum of 7,200 pounds. The representative aircraft requires 28,500 horsepower; therefore, the residual thrust becomes $0.26 \times 28,500$, or 7,400 pounds. Thus it appears that pitch and yaw control gradients which compare to those of a helicopter and are similar to other VTOL designs can be generated using only the residual thrust of the turboshaft engines. The required roll gradient of 0.2 radian/second² can be developed by a 5,100 pound thrust differential between the ducted propellers, which corresponds to only a 5.5 percent change in the minimum thrust required for hovering.

7. CONCLUSIONS AND RECOMMENDATIONS

7.1 Final Configuration Description

The Weight Lifter preliminary design was determined as a result of an analysis of five possible configurations and a subsequent optimization investigation of various parameters for the selected configuration. The final configuration is shown in Figure 19 as consisting of a pod-and-boom fuselage upon which are mounted two ducted propellers and conventional horizontal and vertical tail surfaces.

The ducted propellers are positioned horizontally to provide direct lift for hovering and vertical flight, and may be tilted forward a maximum of 55 degrees to provide forward propulsion as well. The counter-rotating propellers are powered through extension drive shafts by several interconnected turboshaft engines mounted in the top of the fuselage.

The residual thrust of the turboshaft engines is ducted through the fuselage boom to a thrust diverting nozzle at the aft of the boom, and provides the pitch and yaw control forces in hovering. The tail surfaces furnish pitch and yaw control in forward flight, while the residual thrust is utilized to supplement the forward propulsive force of the tilted ducted propellers. Roll control is supplied by differential thrust of the ducted propellers in both hovering and forward flight.

The large fuselage, with a cargo volume two to three times that of existing fixed wing transports, may be loaded through large doors at either end. A helicopter type pilots' compartment is mounted at the nose of the fuselage to provide a maximum field of vision for hovering and vertical flight.

The Weight Lifter design has been optimized at various overall sizes to accommodate the specified range of payload, as shown in the following table:

Payload	7 tons	16 tons
Gross Weight	60,400 lb	135,000 lb
Power Required	18,200 HP	40,800 HP
Ducted Propeller Dia.	18 ft	26.7 ft
Range	180 n.mi.	135 n.mi. (Reference,
Max. Velocity	150 knots	150 knots Figure 27)

The three-view drawing shown in Figure 19 is a representative aircraft, designed to accommodate an average payload of 11 tons.

The optimum disk loading was found to be 150 pounds per square foot, based on a criterion of maximum payload ton-mile per unit gross weight.

A significant feature of the Weight Lifter design is the use of conventional components and construction methods wherever possible, which con-

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tributes to lower cost and increased reliability. An example of this is the conventional variable pitch propellers.

The Weight Lifter requires a relatively small site from which to operate, as evidenced by the representative aircraft shown in Figure 19, which is encompassed in an area with a diameter of approximately 100 feet. The enclosure of the critical whirling propellers in shrouds adds to the suitability of the Weight Lifter for operation in close quarters.

7.2 Further Studies Recommended

A lack of theoretical and experimental information on ducted propellers has necessitated gross assumptions in the calculations involved in this analysis of an optimum Weight Lifter configuration. In order to refine this study and to aid future ducted propeller investigations, it is recommended that both static and wind tunnel testing be conducted in addition to theoretical research on the performance and stability of ducted propellers.

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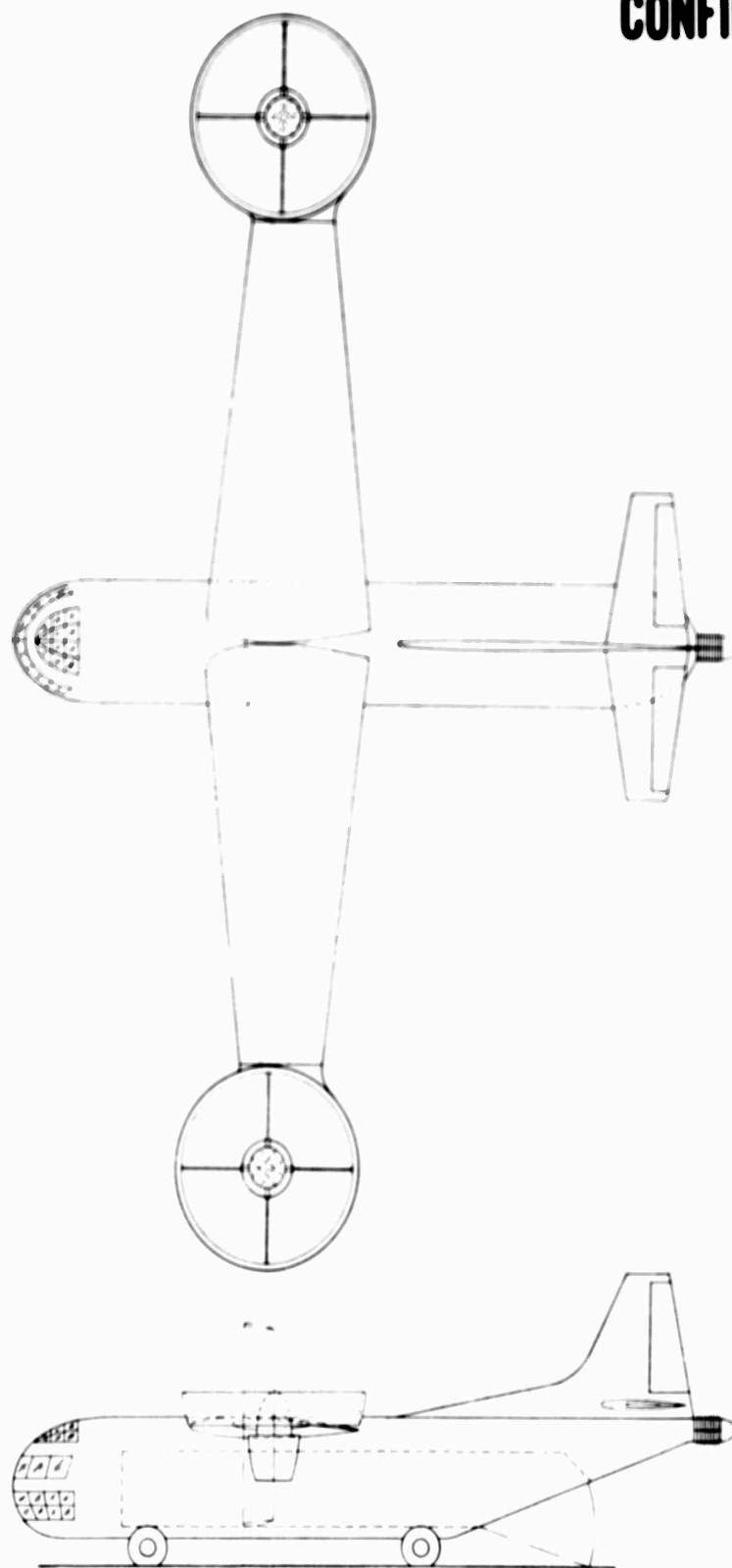
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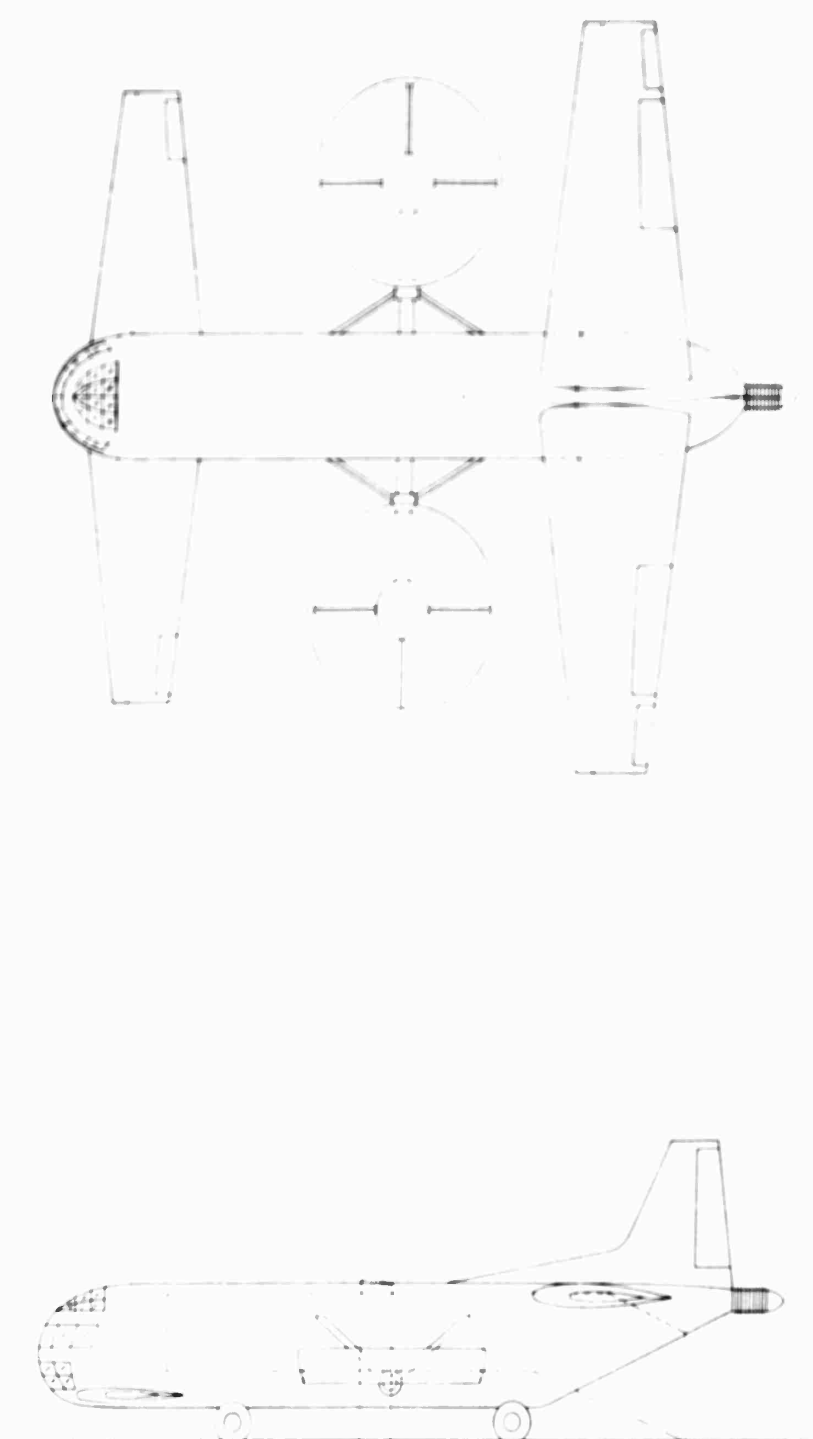
TWO DUCTED PROPELLER
MOUNTED AT TIP OF STRAIGHT WING

SCALE NONE	DRAWN APPV'D	DATE 10 18 56	DRAWING NO.
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FIGURE 1

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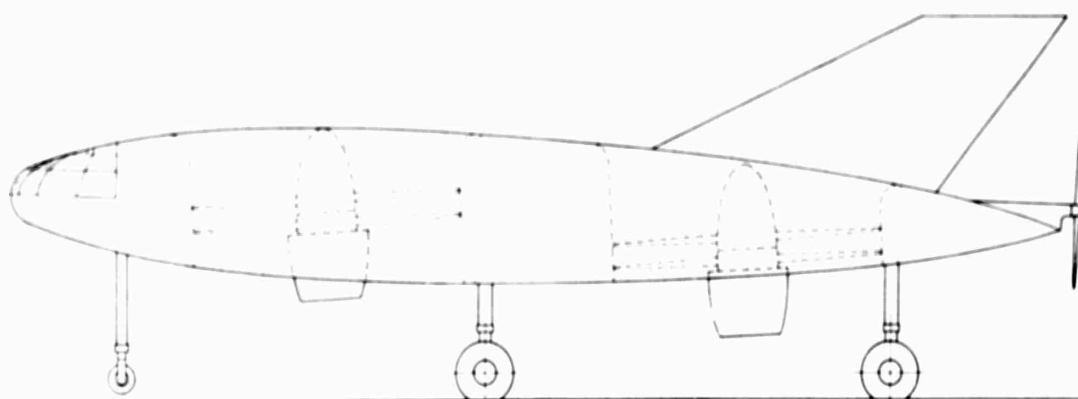
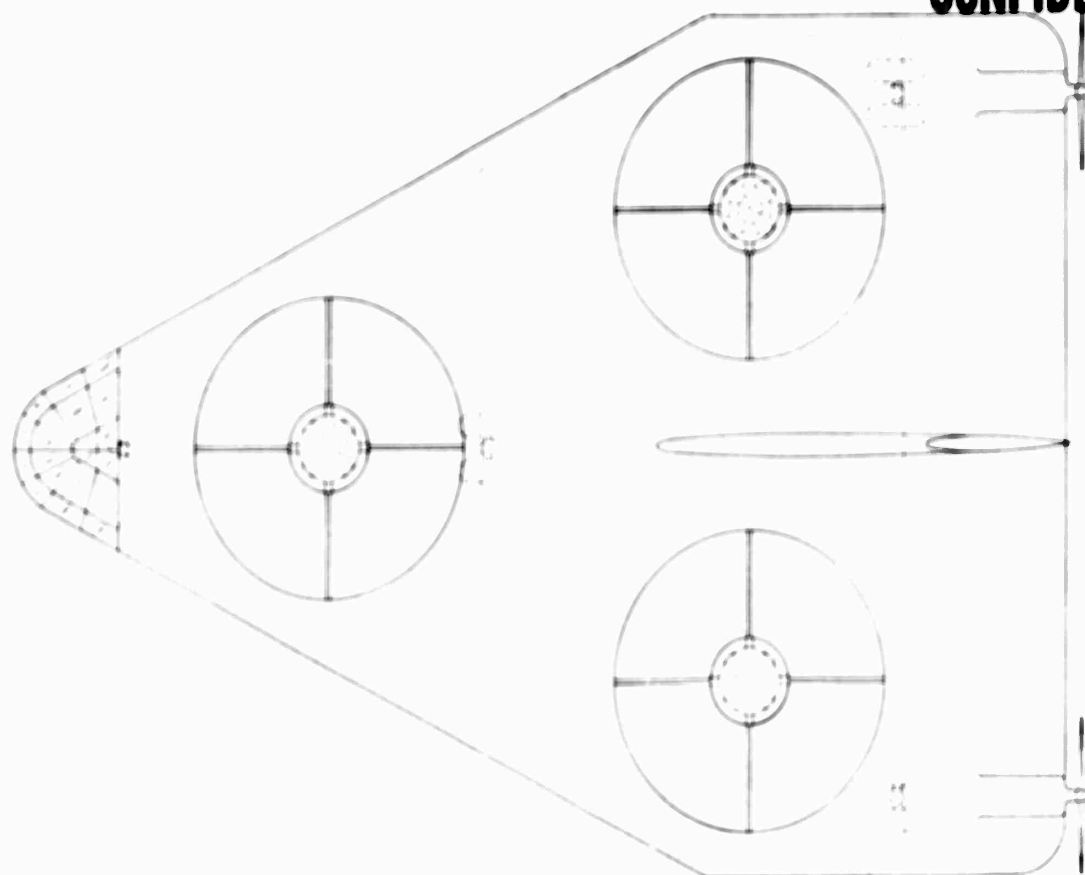
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TWO DUCTED PROPELLER MOUNTED ON FUSELAGE BETWEEN TWO WINGS			
SCALE	DRAWN	DATE	DRAWING NO.
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FIGURE 2

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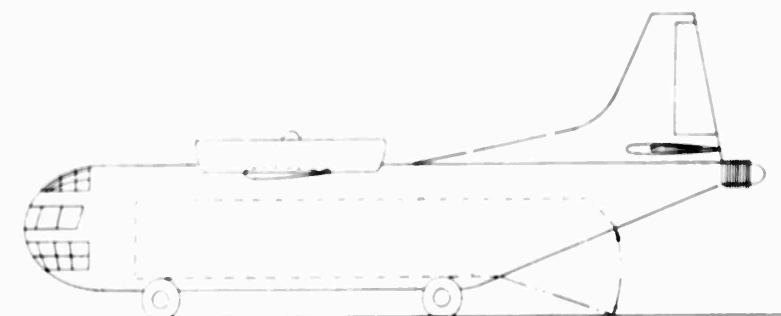
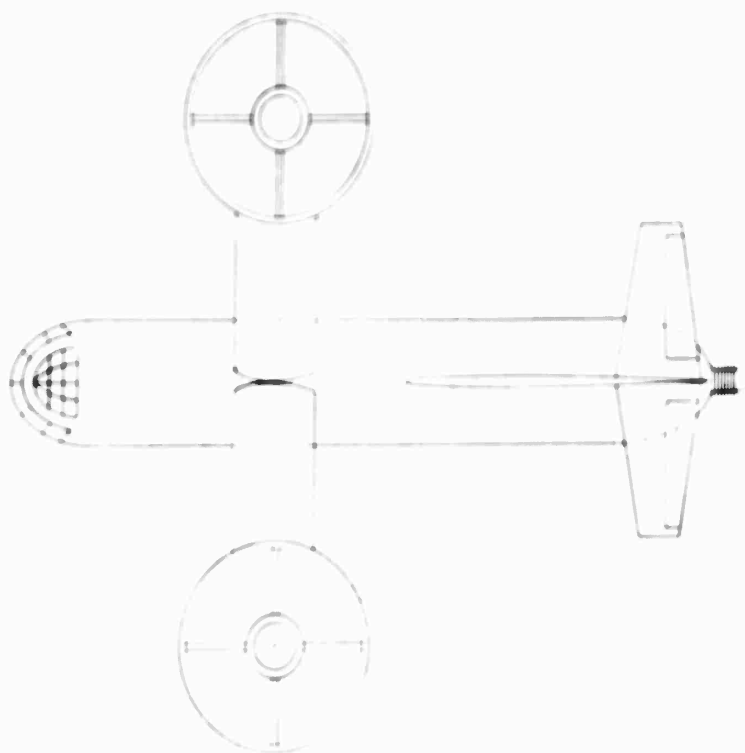
THREE DUCTED PROPELLER
BUILT IN DELTA WING

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FIGURE 3

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TWO DUCTED PROPELLER MOUNTED ON FUSELAGE			
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FIGURE 4

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FOUR DUCTED PROPELLER MOUNTED ON FUSELAGE			
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FIGURE 5

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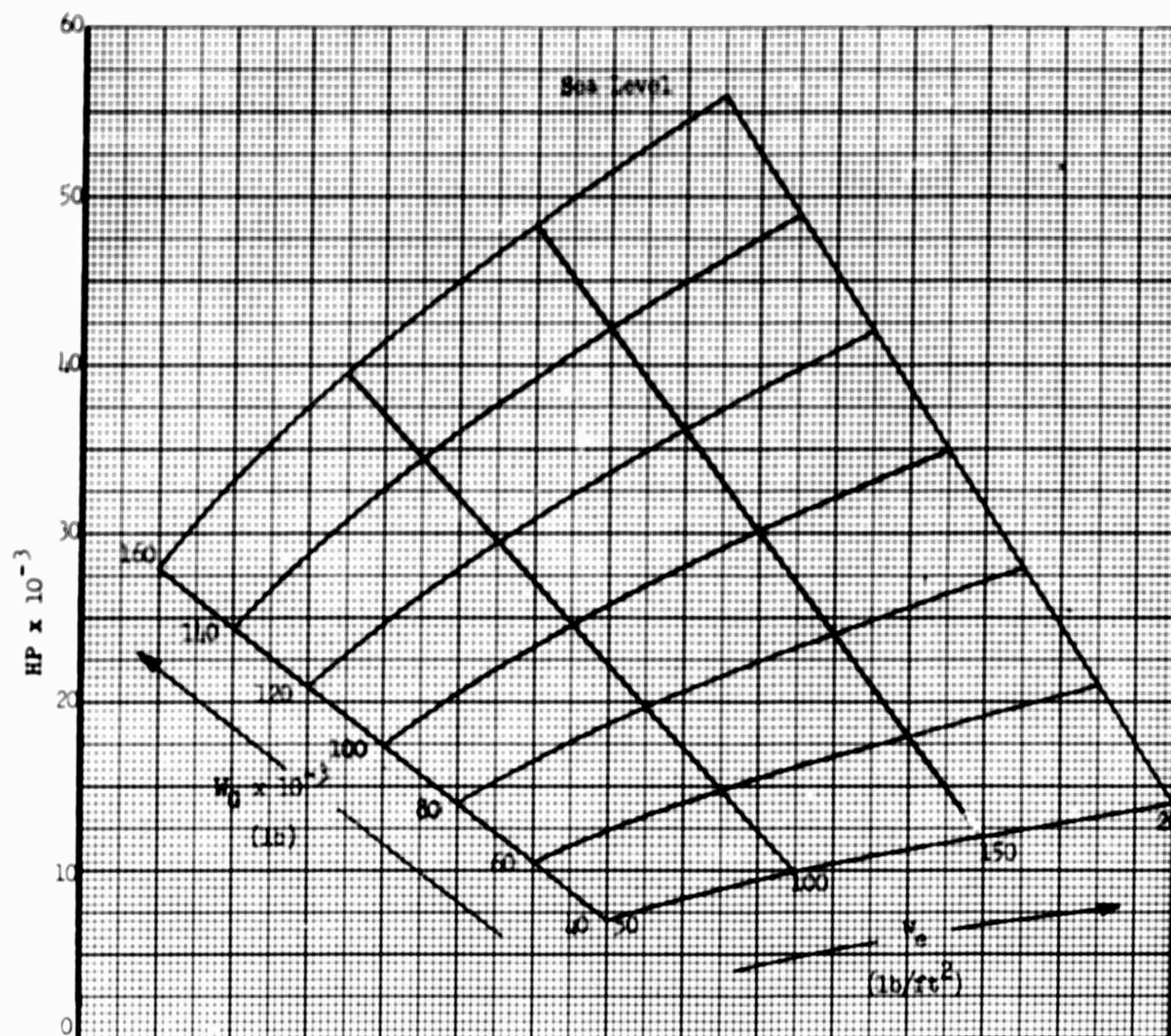


FIGURE 6: HORSEPOWER REQUIRED FOR HOVERING + 20% FOR CLIMB

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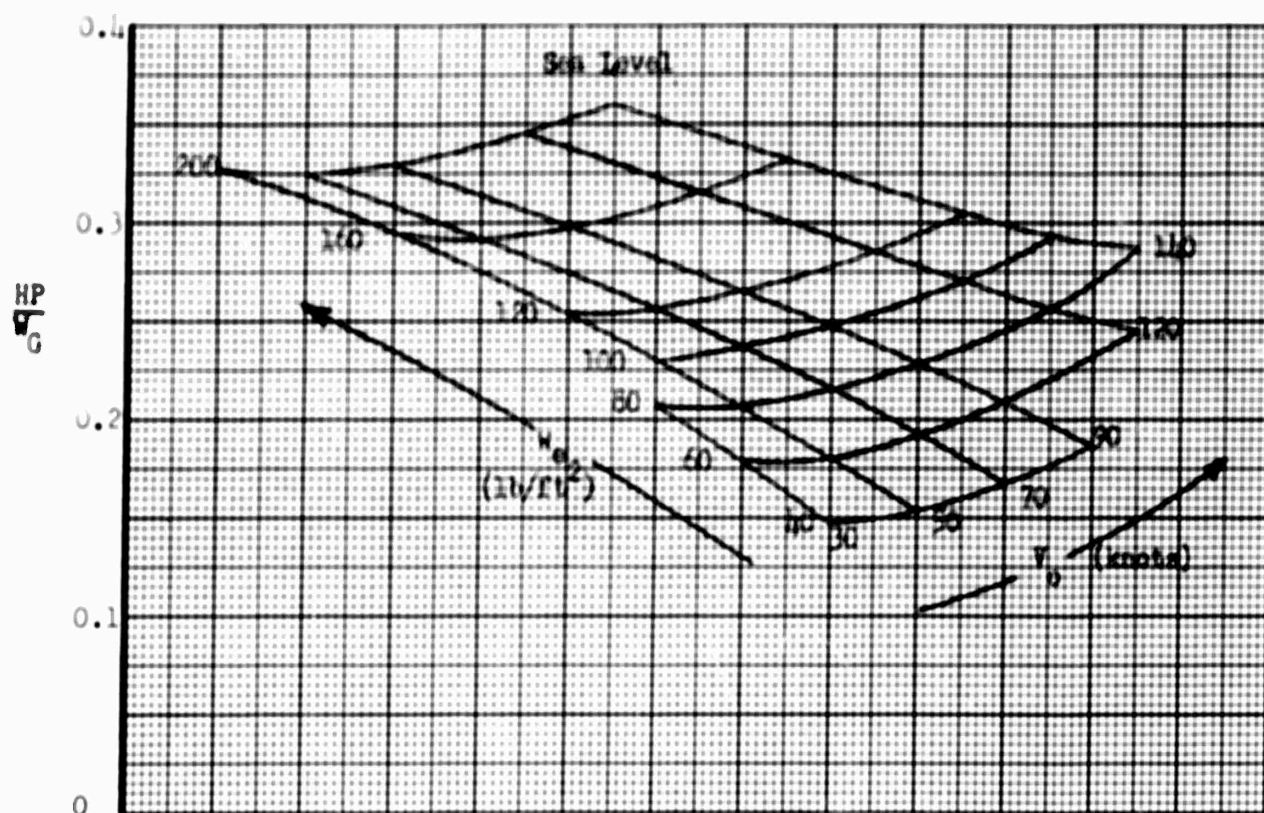


FIGURE 8: HORSEPOWER REQUIRED FOR LEVEL FLIGHT; TILTING DUCT

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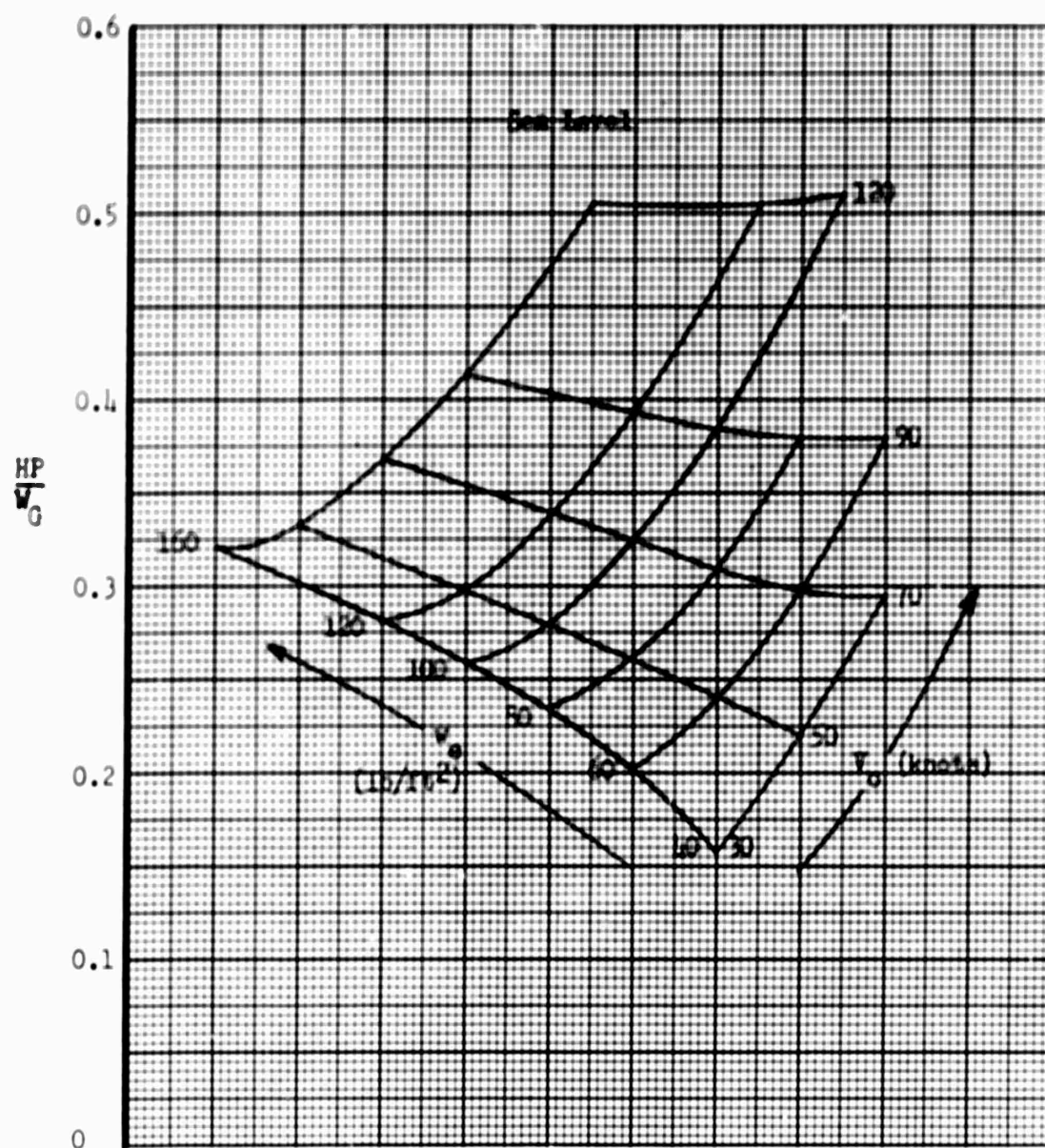


FIGURE 9: HORSEPOWER REQUIRED FOR LEVEL FLIGHT;
FIXED DUCTS AND PUSHER PROPELLERS

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$P = 22,000 \text{ lb}$

Optimum Result:
 $W_G = 79,000 \text{ lb}$
 $w_e = 50 \text{ lb/ft}^2$

— R_F Available
 ▲ R_F Required
 + Optimum

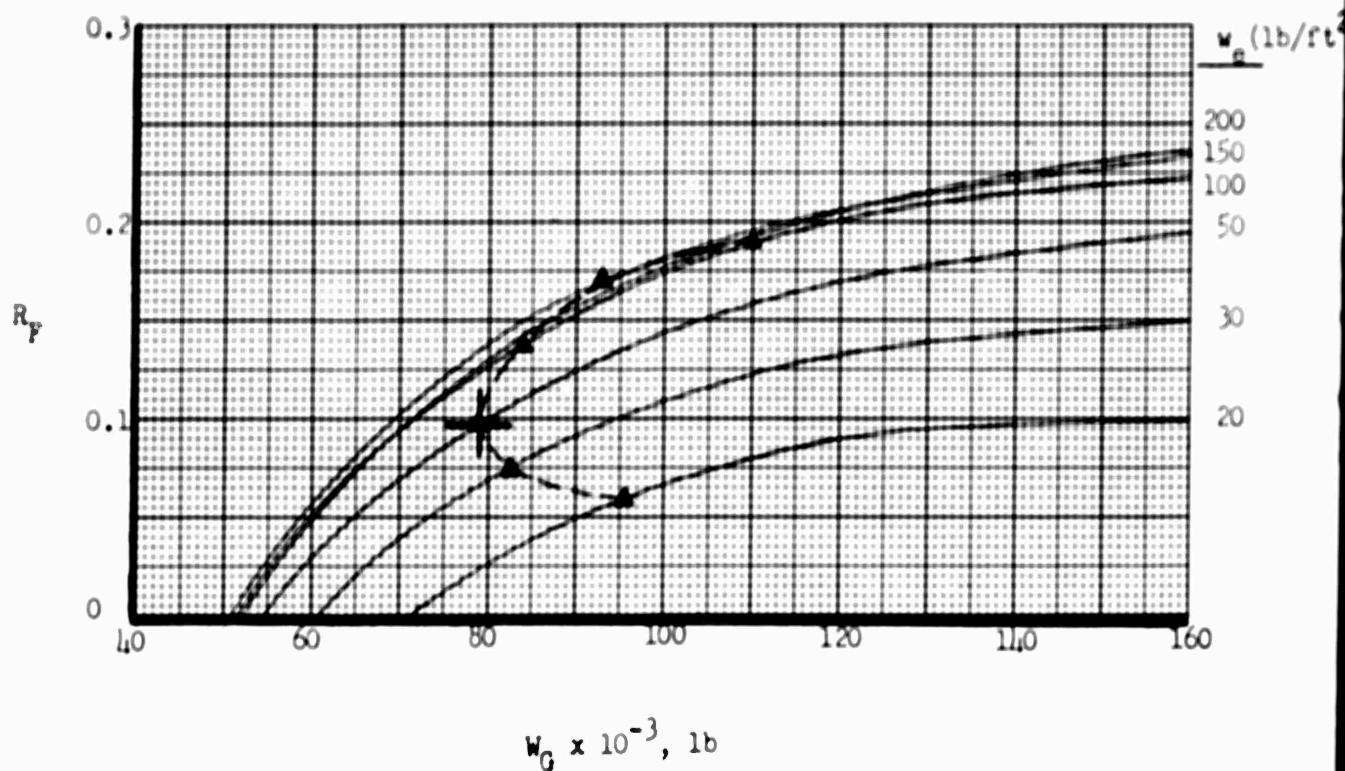


FIGURE 10: R_F GRAPHICAL METHOD OPTIMUM DESIGN FOR TWO-DUCT, NO-WING WEIGHT LIFTER

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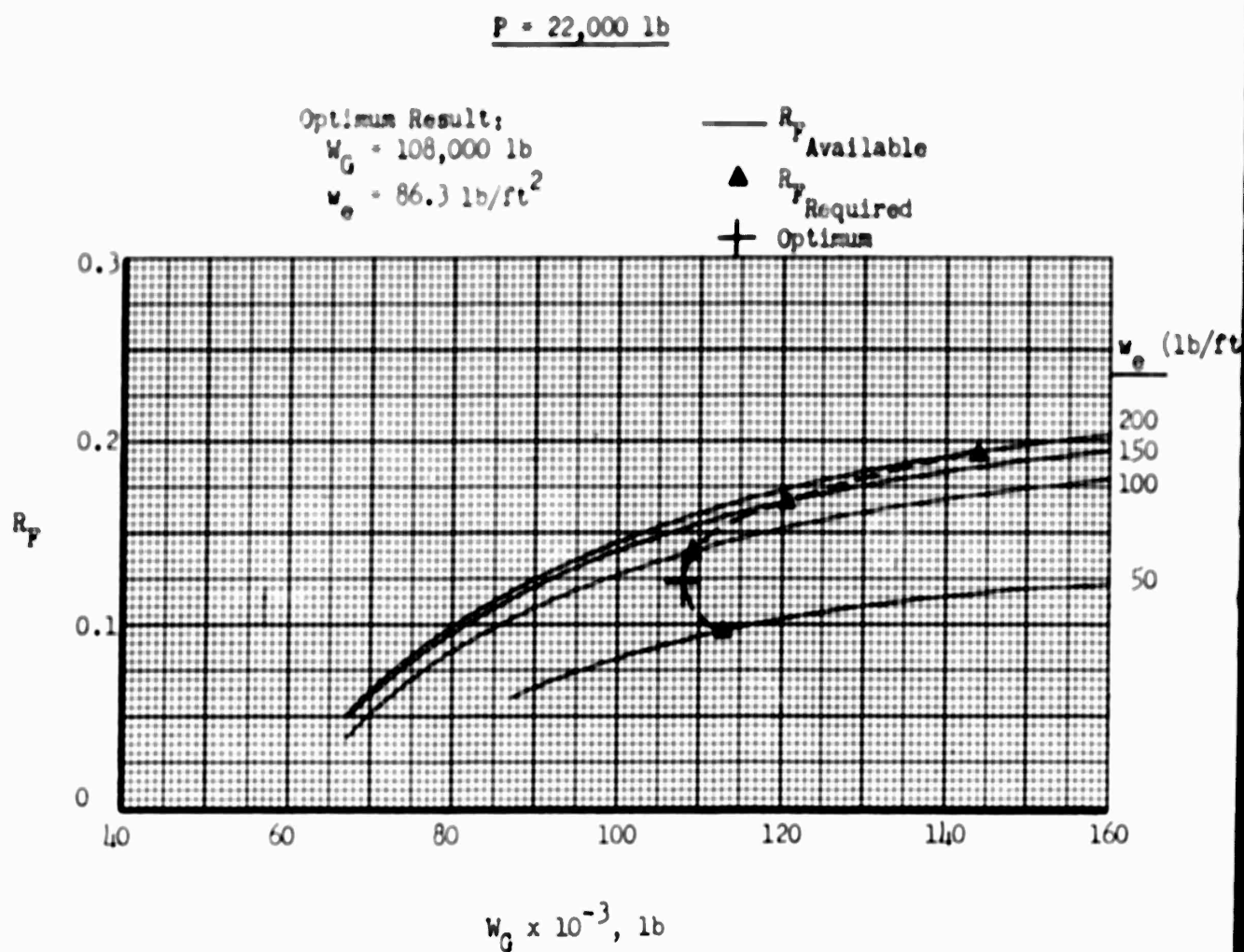


FIGURE 11: R_F GRAPHICAL METHOD OPTIMUM DESIGN FOR FOUR-DUCT WEIGHT LIFTER

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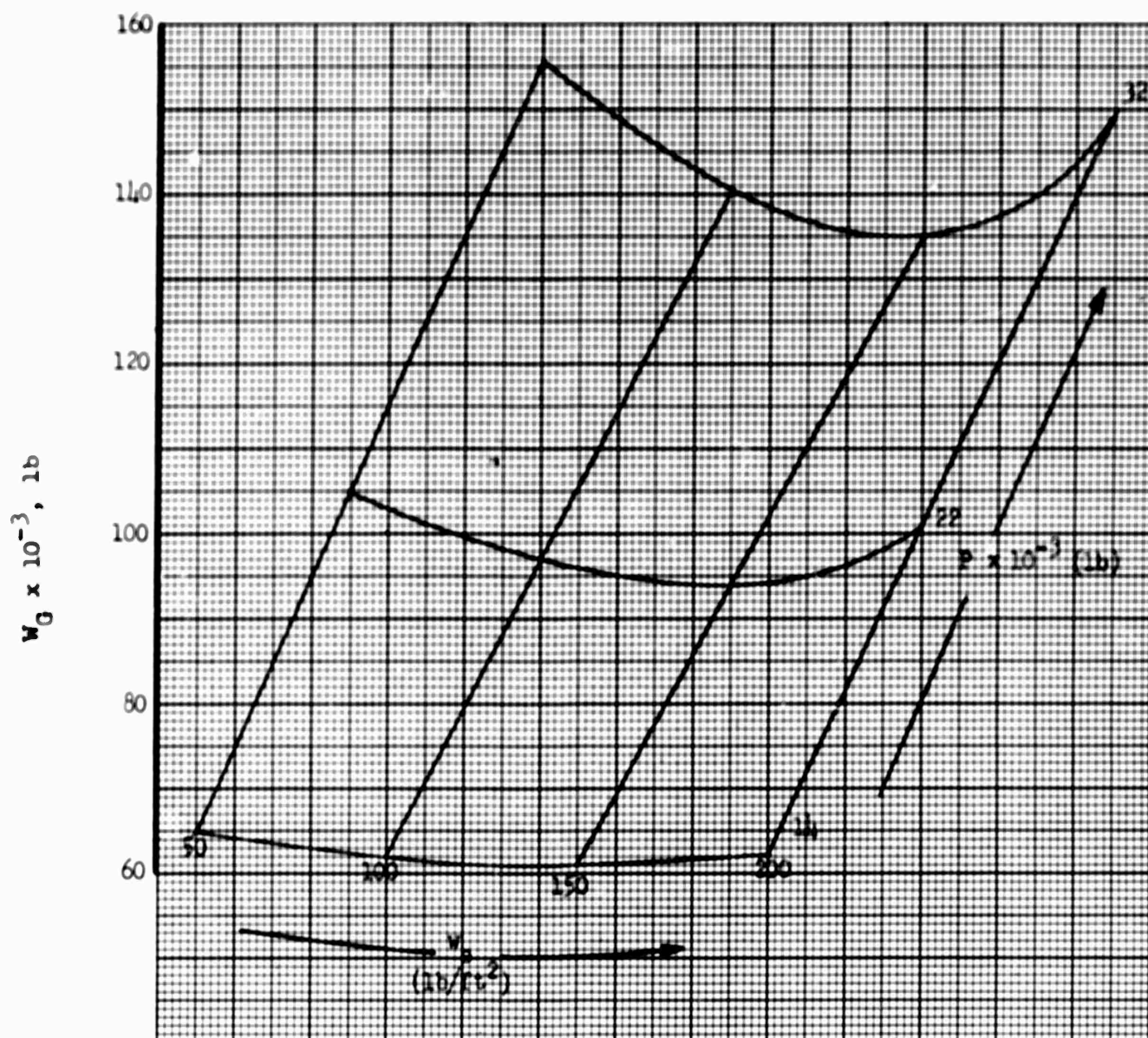


FIGURE 12: DISK LOADING FOR MAXIMUM R/W_G

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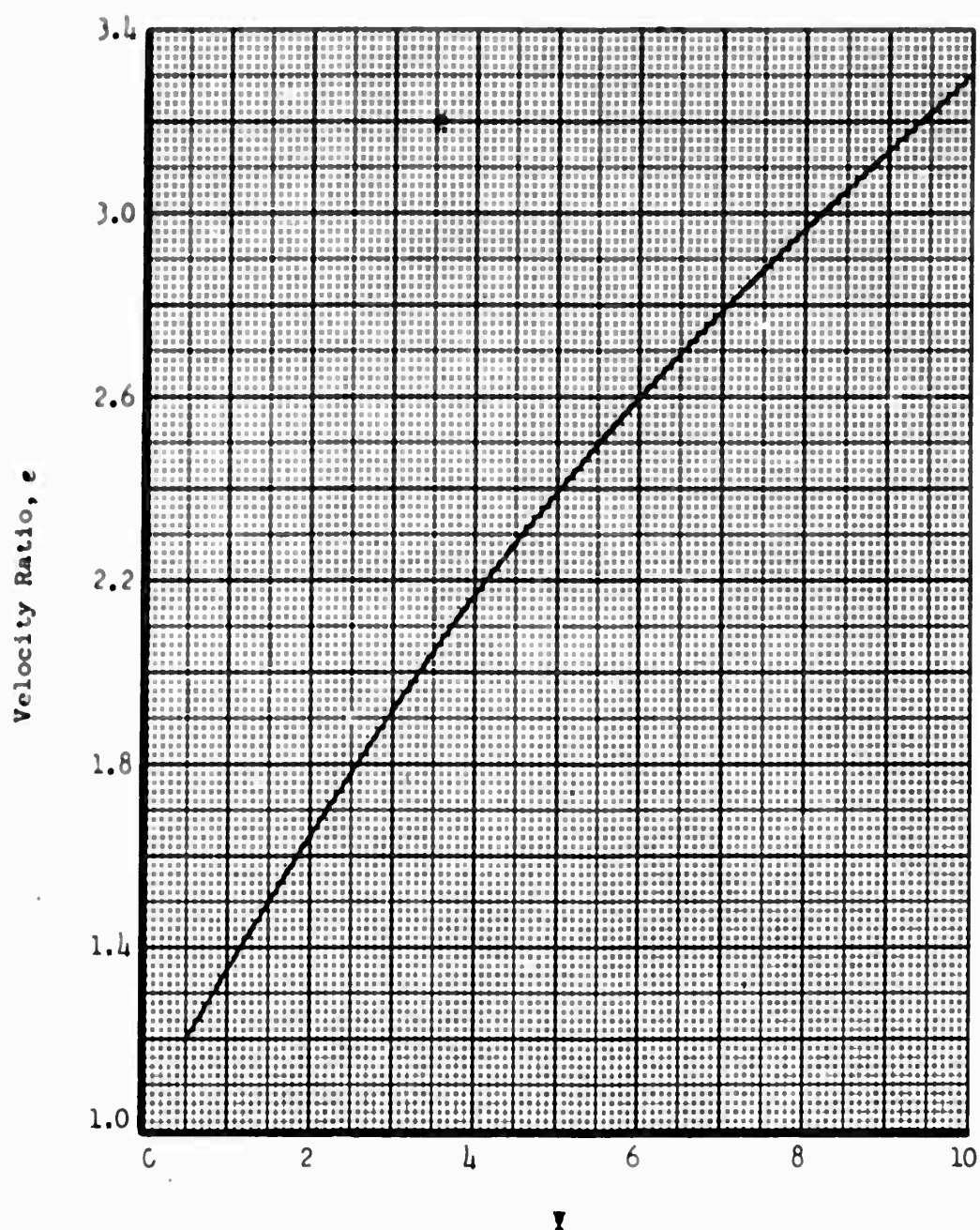


FIGURE 13: DUCT EXIT TO FREE STREAM VELOCITY RATIO FOR TILTING DUCTS + RESIDUAL THRUST

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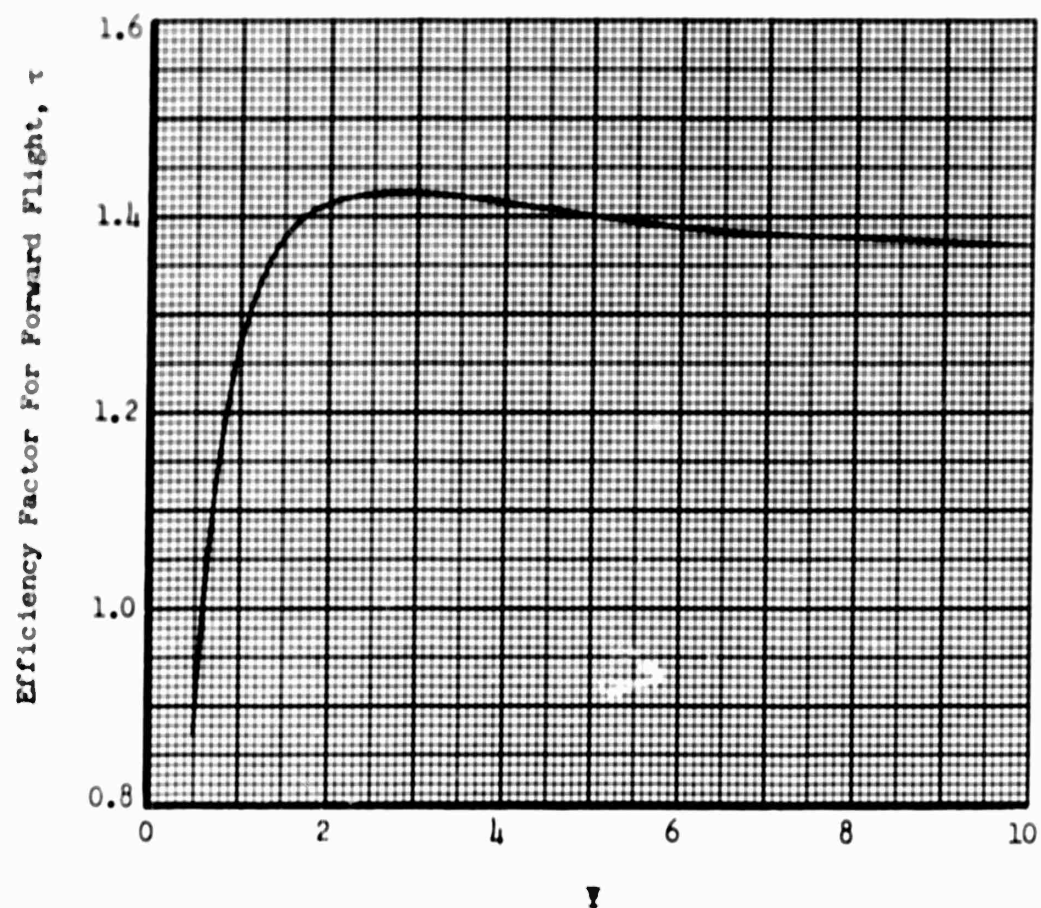


FIGURE 14: τ FOR TILTING DUCTS + RESIDUAL THRUST

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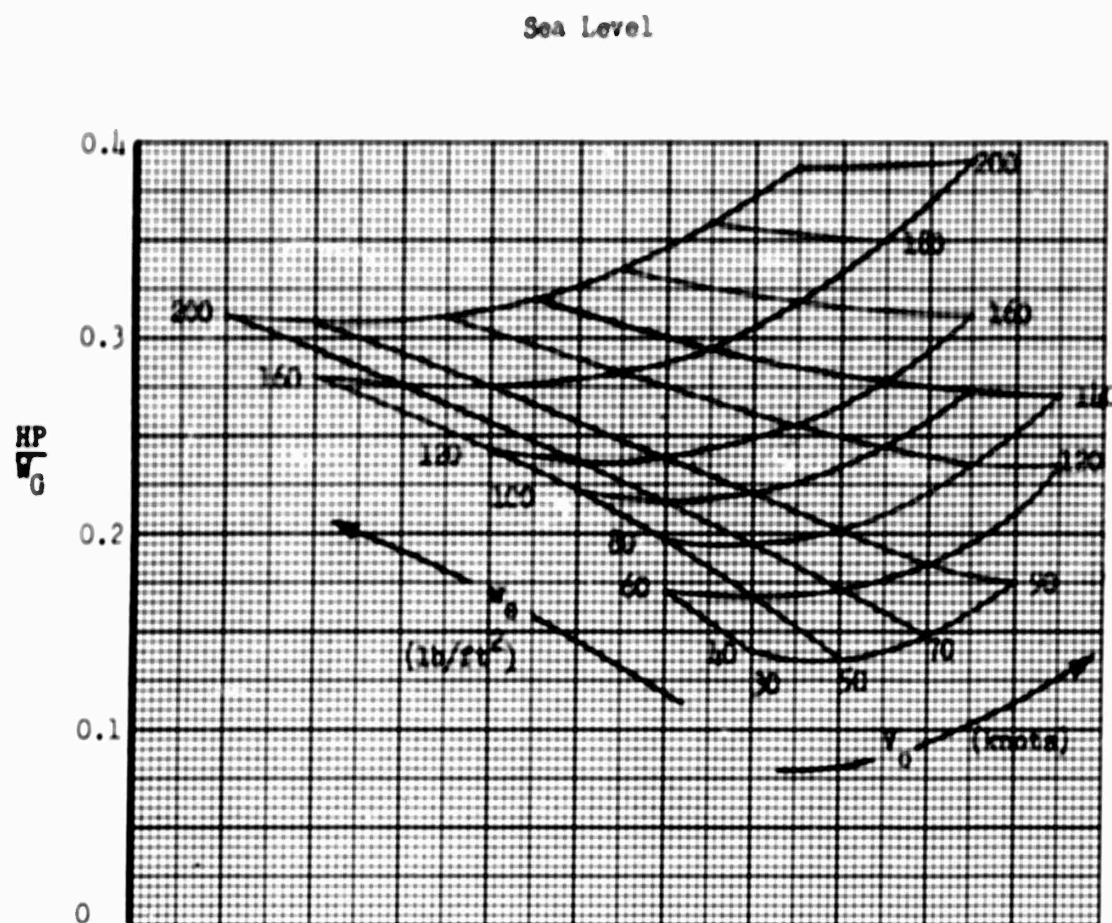


FIGURE 15: HORSEPOWER REQUIRED FOR LEVEL FLIGHT; TILTING DUCTS + RESIDUAL THRUST

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$$C_{D_1} = 0.10$$

$$C_{D_e} = 0.24$$

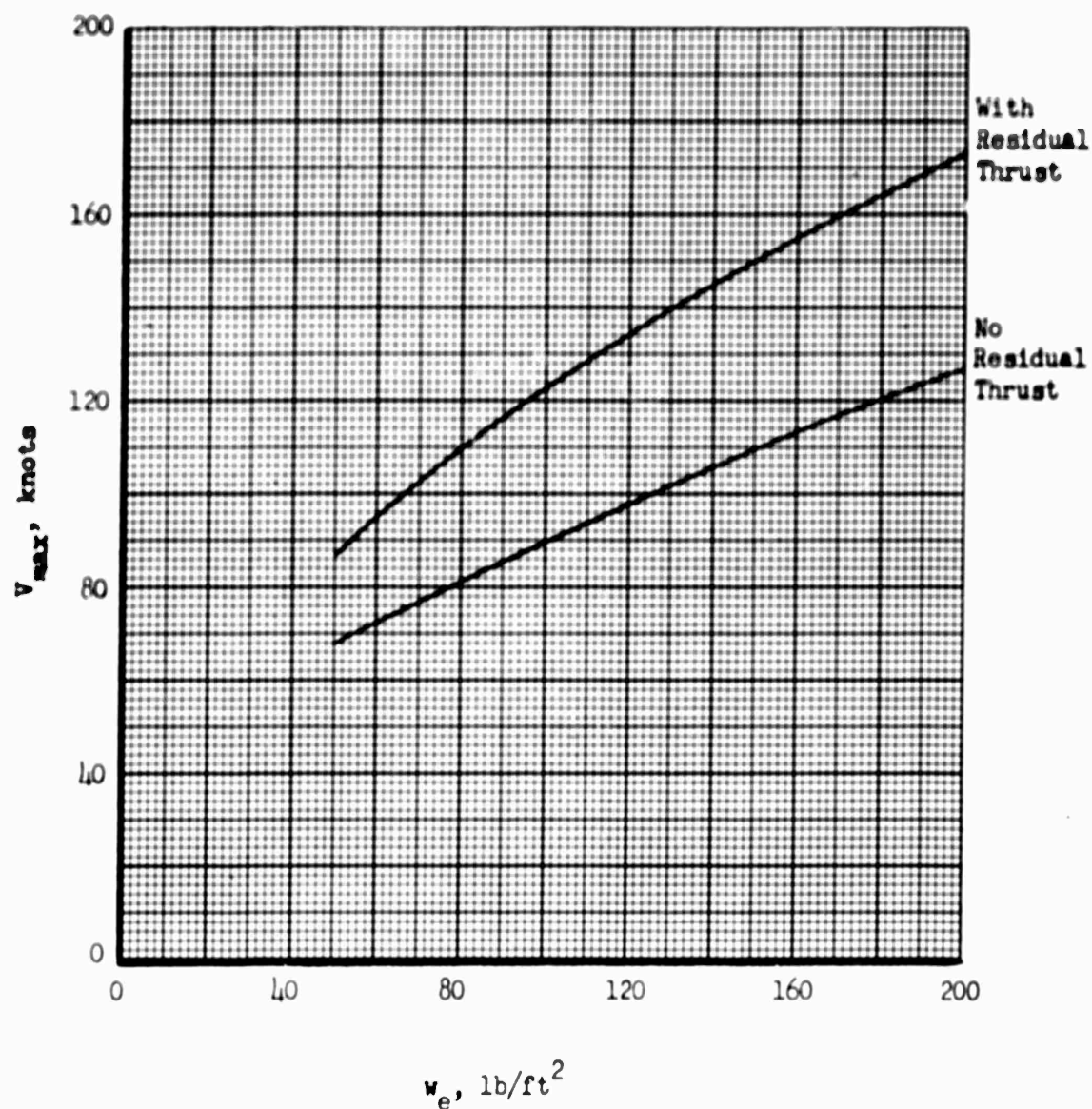


FIGURE 16: OPTIMUM WEIGHT LIFTER MAXIMUM LEVEL FLIGHT VELOCITY

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$$w_0 = 150 \text{ lb/ft}^2$$

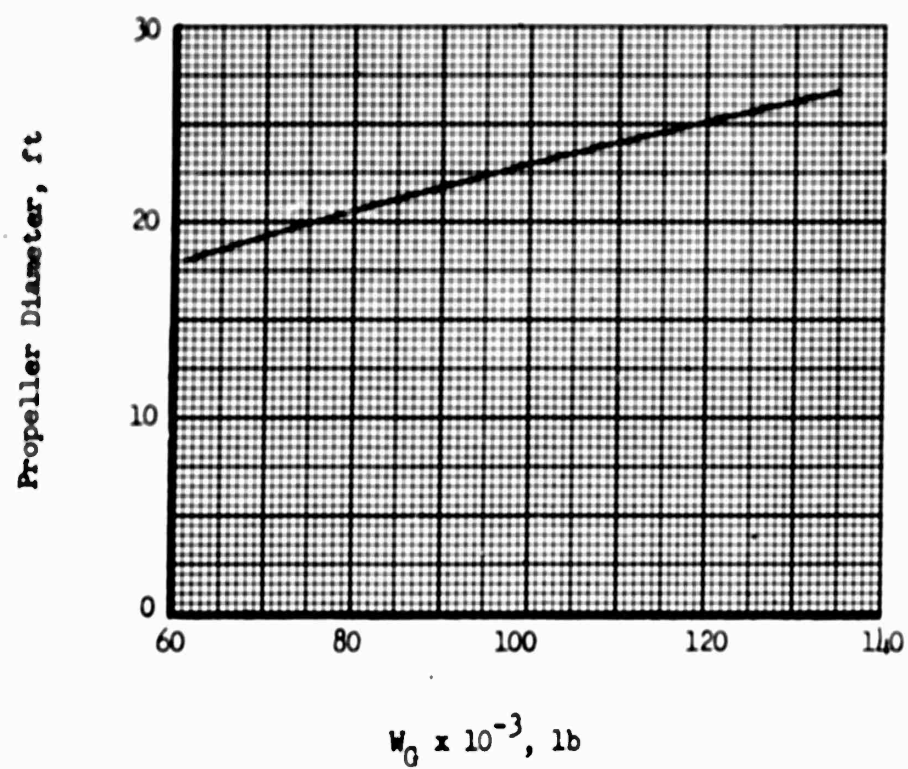


FIGURE 17: REQUIRED PROPELLER DIAMETER

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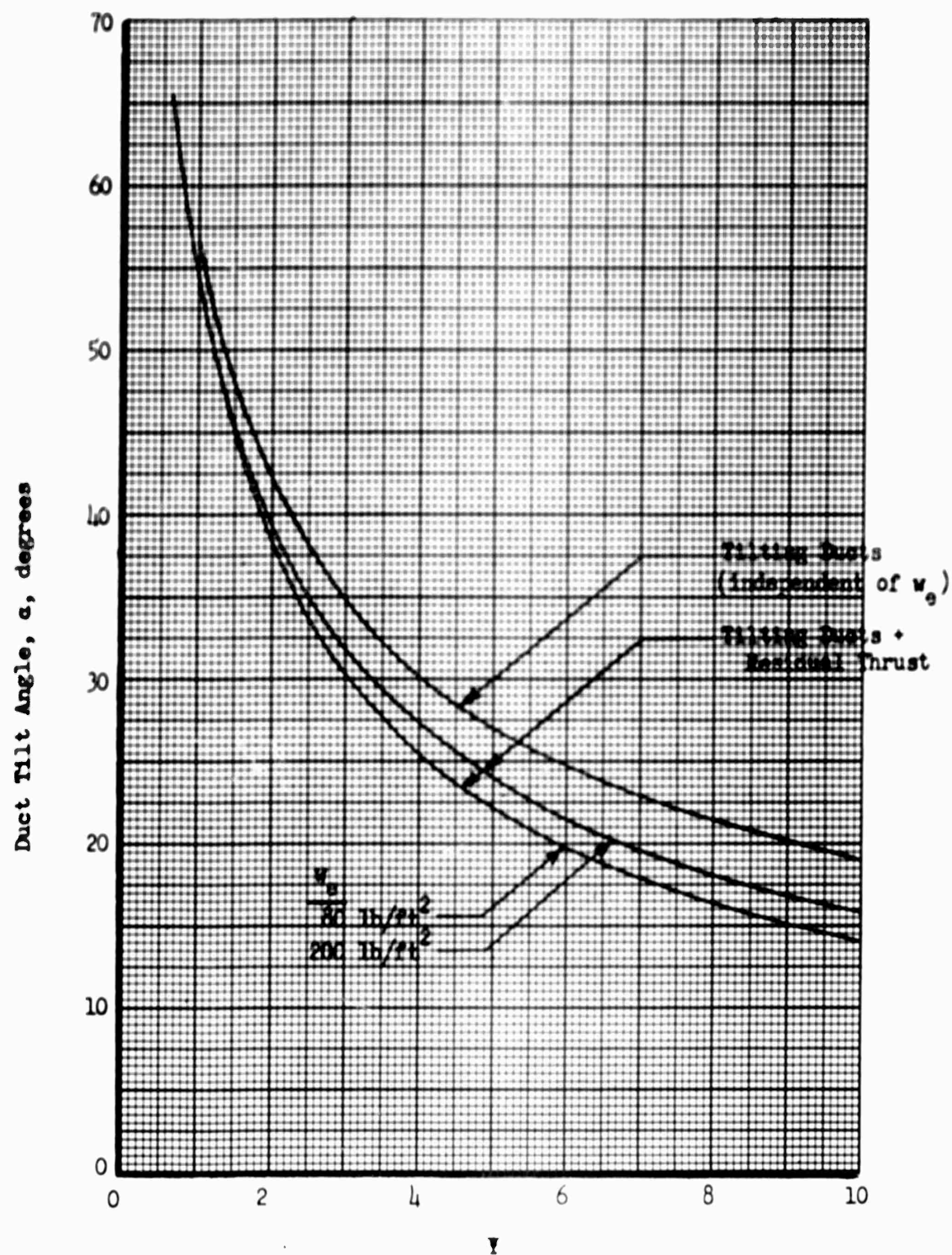
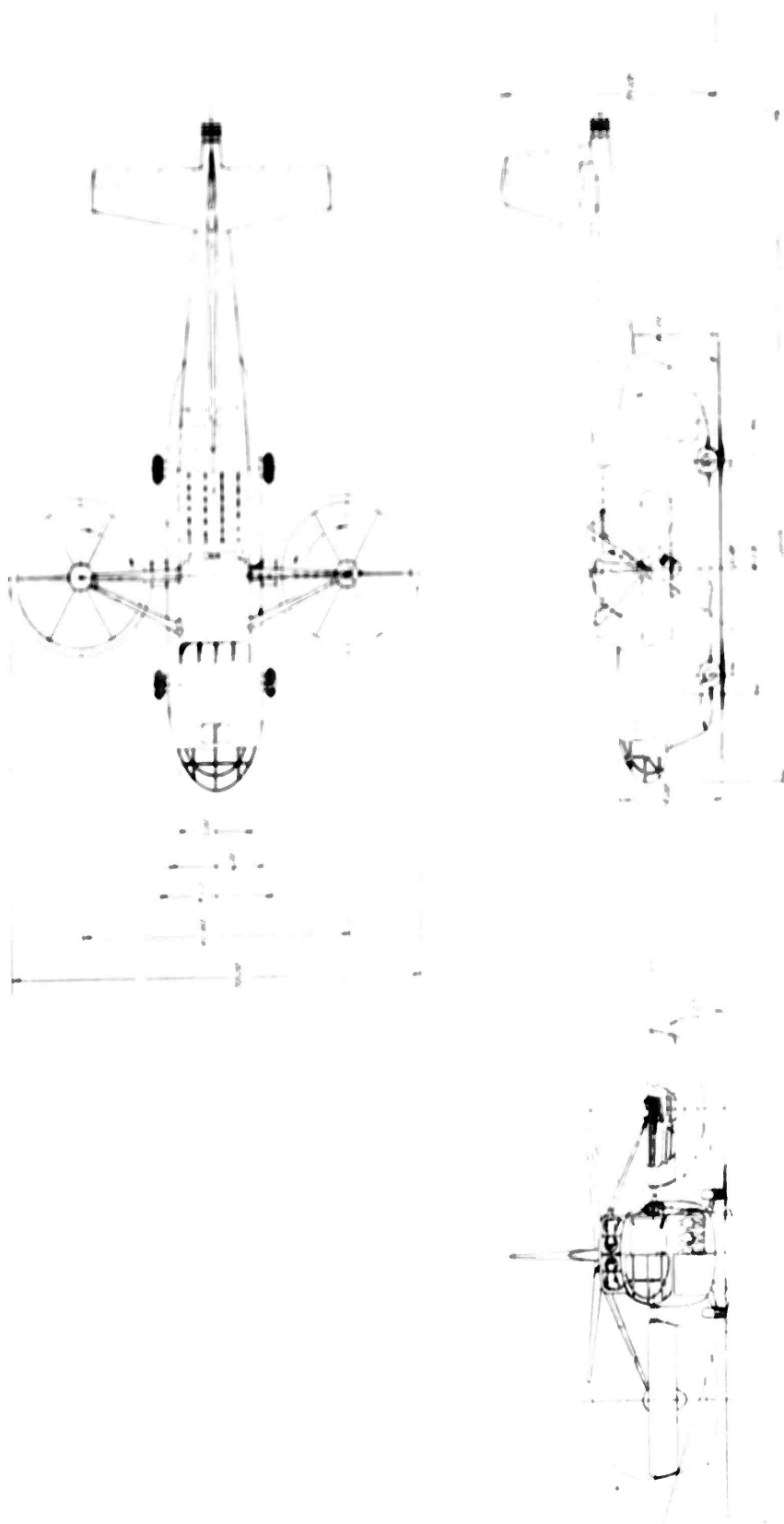


FIGURE 18: DUCT TILT ANGLE FOR LEVEL FLIGHT

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FIGURE 19

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PROVIDES THE HELICOPTER
MOUNTED ON AIRCRAFT

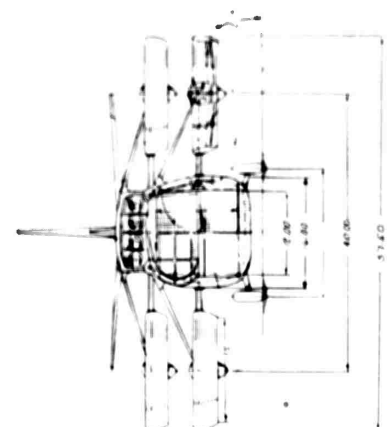
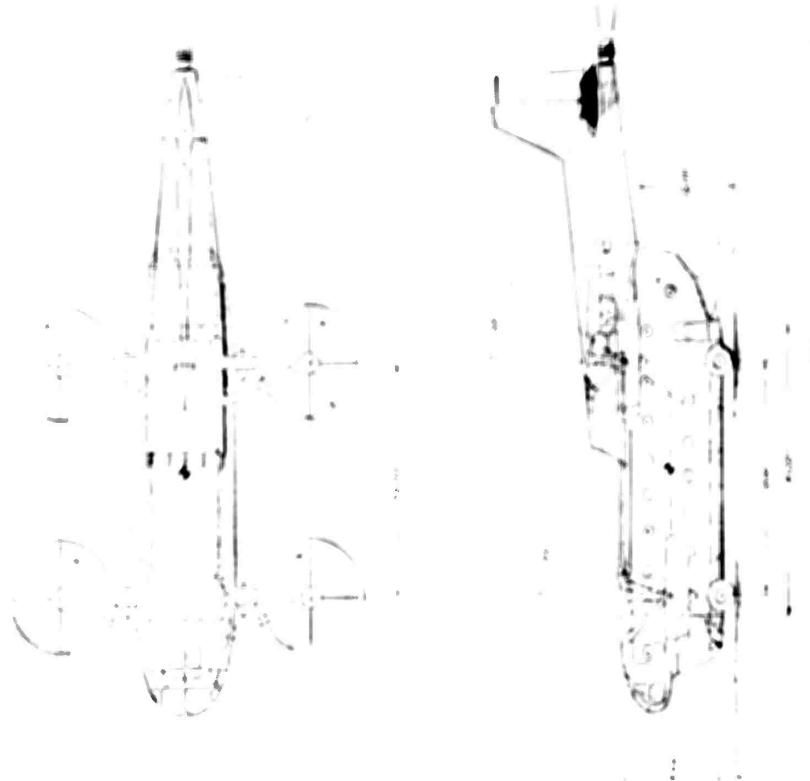


FIGURE 20

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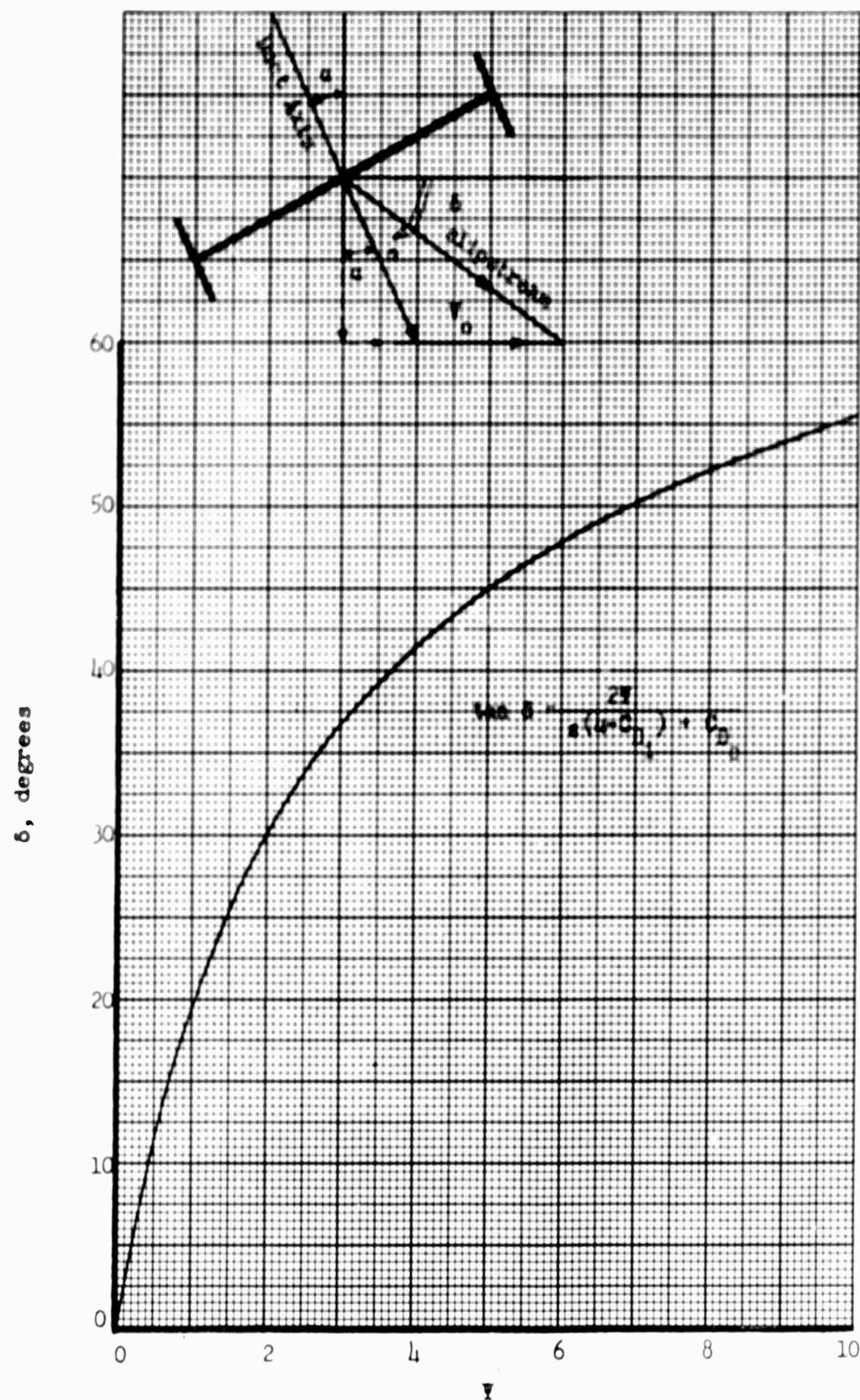


FIGURE 21: SLIPSTREAM DEFLECTION OF A DUCTED PROPELLER IN FORWARD FLIGHT

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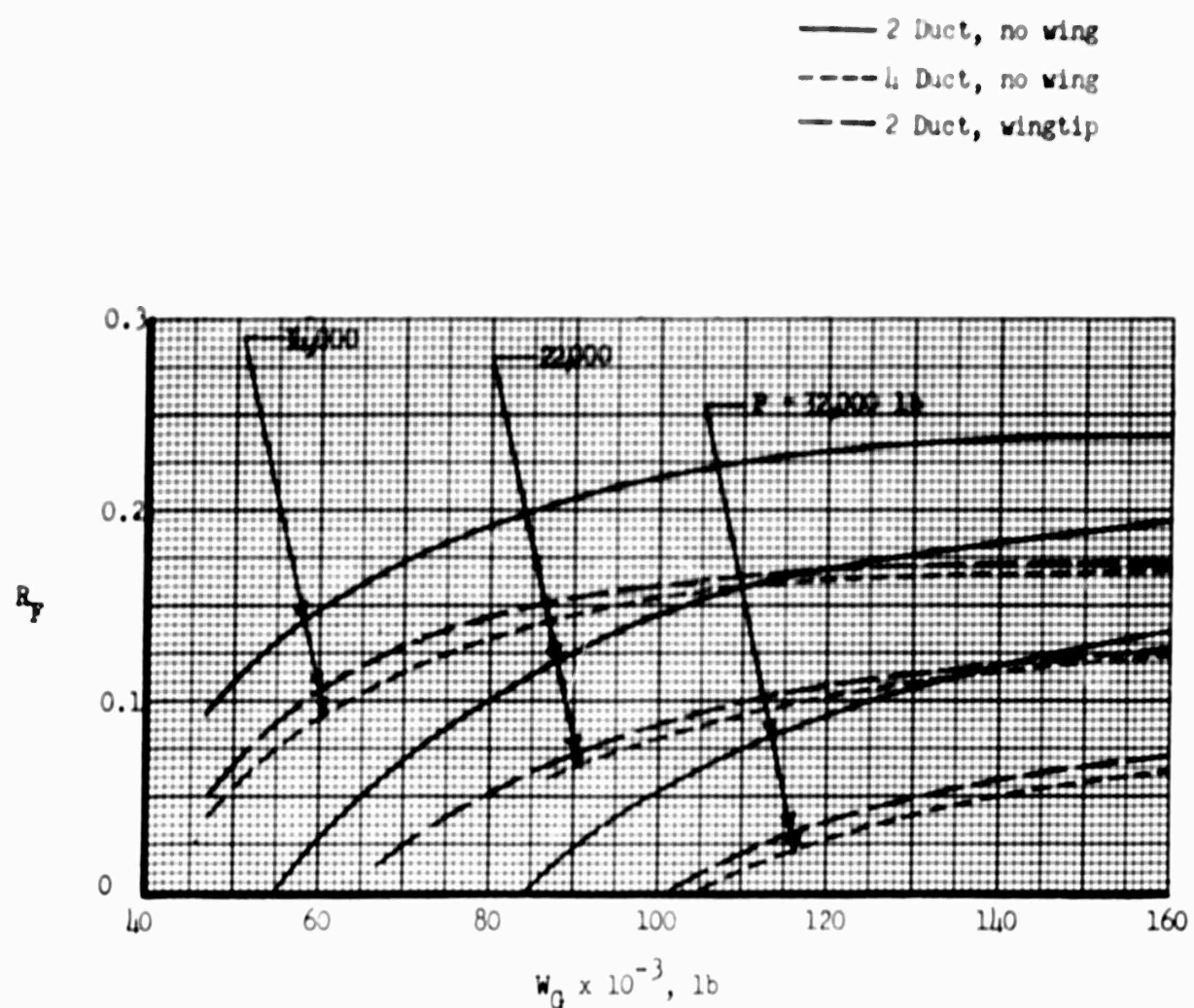


FIGURE 22: R_F AVAILABLE FOR VARIOUS AIRCRAFT CONFIGURATIONS;
EFFECTIVE DISK LOADING = 50 lb/ft²

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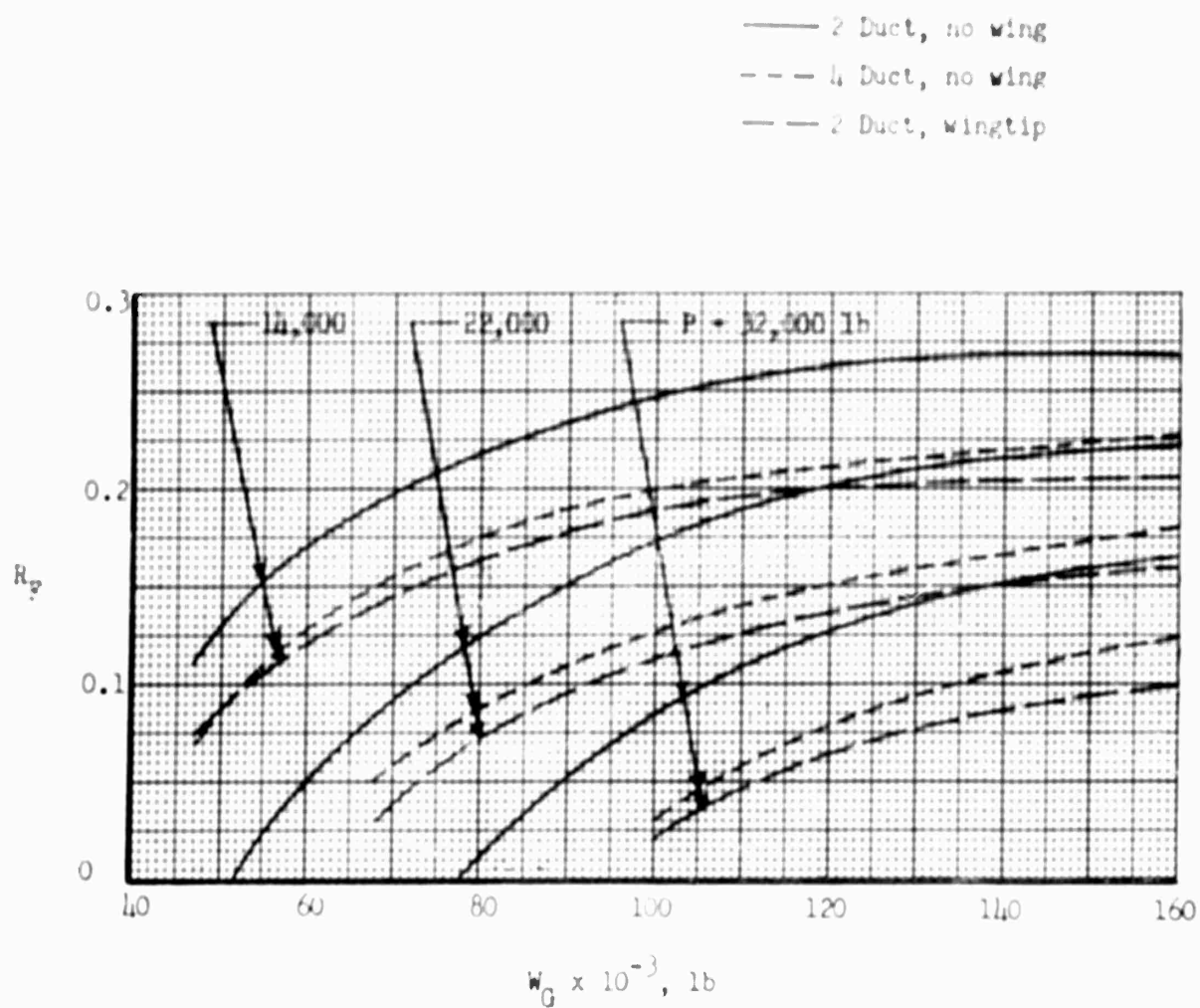


FIGURE 23: R_F AVAILABLE FOR VARIOUS AIRCRAFT CONFIGURATIONS;
EFFECTIVE DISK LOADING = 100 lb/ft^2

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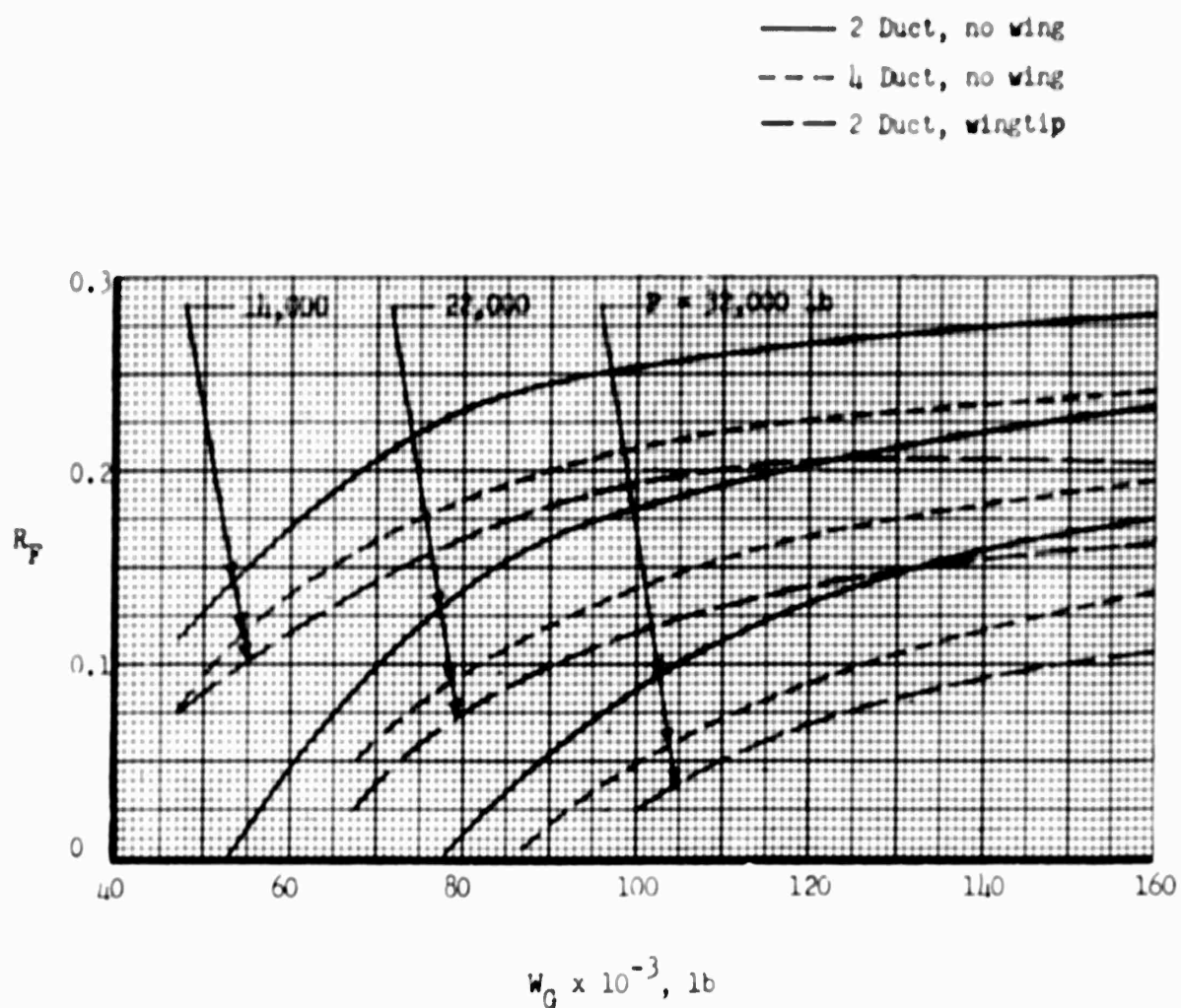


FIGURE 24: R_F AVAILABLE FOR VARIOUS AIRCRAFT CONFIGURATIONS;
EFFECTIVE DISK LOADING = 150 lb/ft^2

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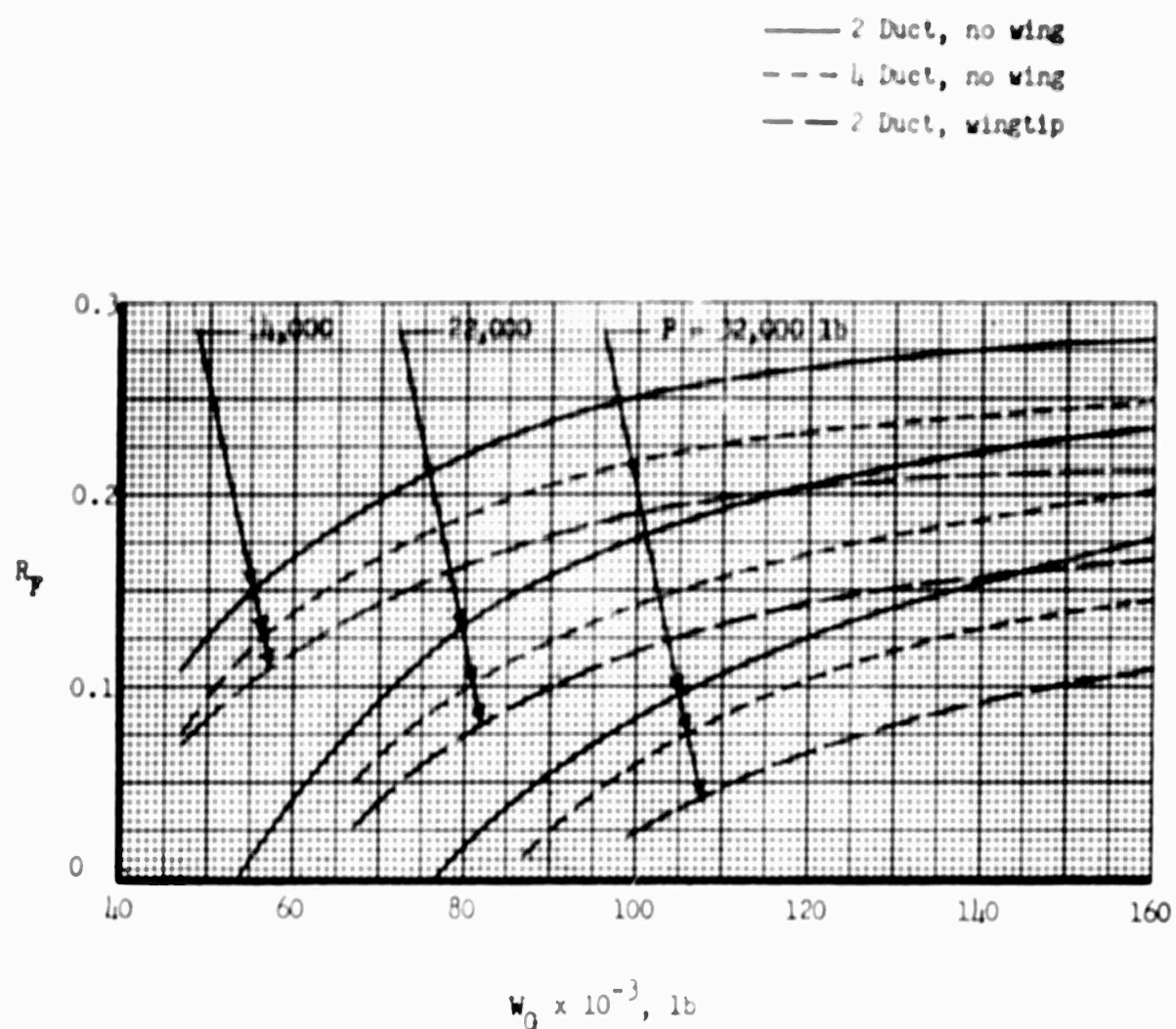


FIGURE 25: R_F AVAILABLE FOR VARIOUS AIRCRAFT CONFIGURATIONS,
EFFECTIVE DISK LOADING = 200 lb/ft²

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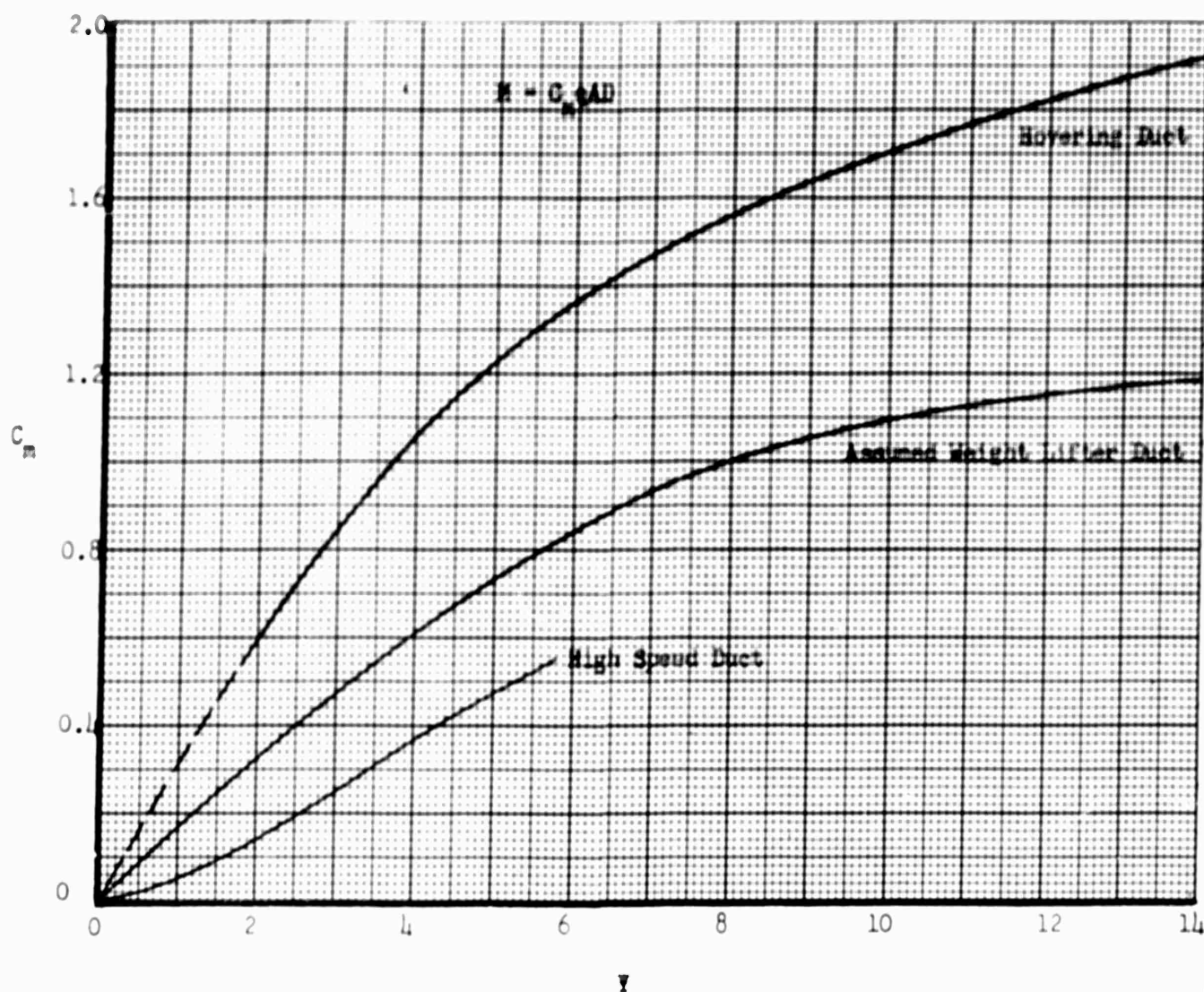


FIGURE 26: DUCTED PROPELLER PITCHING MOMENT COEFFICIENT

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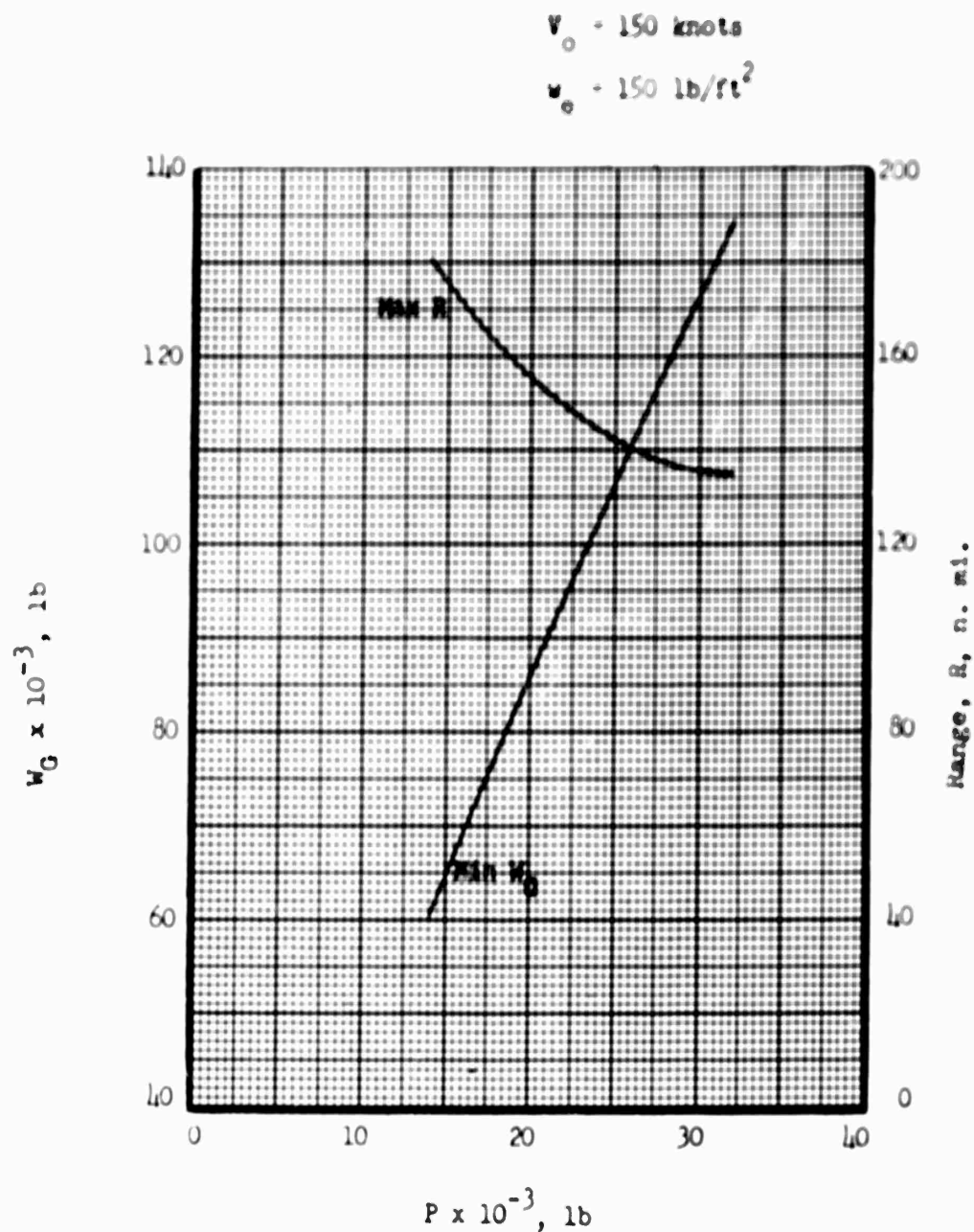


FIGURE 27: VARIATION OF GROSS WEIGHT AND RANGE WITH PAYLOAD FOR OPTIMUM DESIGN TWO-DUCT WEIGHT LIFTER

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