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AFFDL-TR-70-44

CONFIGURATION DESIGN ANALYSIS OF A PROP/ROTOR AIRCRAFT

DAVID A. RICHARDSON,
JAAN LIIVA, et al
The Boeing Company

TECHNICAL REPORT AFFDL-TR-70-44

APRIL 1970

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CONFIGURATION DESIGN ANALYSIS OF A PROP/ROTOR AIRCRAFT

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The Boeing Company

FOREWORD

This report was prepared by the Boeing Company, Vertol Division of Philadelphia, Pennsylvania for the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio under contract F33615-69-C-1570, project No. 698BT, "US/FRG V/STOL Technology Program". This contract is for a multiphase effort of parametric studies, detail design, model tests and analysis. This report only covers phase I, configuration design analysis. The results of the other phases will be treated in future reports.

The contract was administered by the Air Force Flight Dynamics Laboratory with Mr. Daniel E. Fraga (FDV) as project engineer. The principal investigators for the Boeing Company were Mr. David A. Richardson and Mr. Jaan Liiva. This report covers the phase I work conducted from 15 April 1969 to 15 August 1969. The final report was submitted by the authors in November 1969.

This report has been reviewed and is approved.

JAY D. PINSON
Lt. Colonel, USAF
Chief, V/STOL Technology Division

ABSTRACT

Basic design studies on tilt prop/rotor aircraft performed as the first phase of the four phase USAF Contract F33615-69-C-1570 are summarized in this interim report. This program is to determine design criteria and demonstrate the adequacy of technology by designing a full-scale prop/rotor aircraft and by designing, manufacturing and testing scaled models. The work reported herein consists of the definition of a prop/rotor preliminary design and performance sensitivity trade-offs. A prop/rotor aircraft which can perform a transport mission with a 250 nautical miles radius, a cruise speed of 350 knots and a payload of five tons with a vertical take-off at 2,500 ft. and 93°F is defined. This aircraft also can perform a rescue mission with a 500 nautical mile radius and a mid-point hover time of thirty minutes. Landing gear sized to provide a coverage of 40 and 38 passes when operated on CBR4 soil is included in this design. A 21 percent wing thickness is used to provide the largest depth of wing compatible with high speed drag rise in order to satisfy the structural requirements of a prop/rotor aircraft with a minimum weight wing. The prop/rotor utilized has no flap or lag hinges. Rotor blade cyclic pitch is planned to provide both control moments and load alleviation. A hover figure of merit of 75 percent and a cruise efficiency of 78 percent are expected to be achieved with this aircraft. Weight estimates based on a fairly conservative projection of technology indicate that the useful load fraction of this aircraft is 31.6 percent.

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LIST OF SYMBOLS

A.F.	activity factor
b	number of blades
c	blade chord at .75R
$\bar{\bar{C}}$	mean aerodynamic chord
c.g.	aircraft center of gravity location, per cent MAC
C_D	aircraft drag coefficient, $\frac{D}{qs}$
C_L	aircraft lift coefficient, $\frac{L}{qs}$
C_{L_I}	integrated design lift coefficient (of a propeller)
$C_{L_{MAX}}$	aircraft maximum lift coefficient
C_{L_p}	propeller force coefficient normal to remote velocity, $\frac{L_p}{\rho n^2 D^4}$
C_M	aircraft pitching moment coefficient, $\frac{M}{qs}$
C_N	propeller normal force coefficient $\frac{N_p}{\rho n^2 D^4}$
C_n	directional stability derivative, rad $^{-1}$
C_p	propeller power coefficient, $\frac{P}{\rho n^3 D^5}$
C_T	propeller thrust coefficient, $\frac{T}{\rho n^2 D^4}$
C_{X_p}	propeller force coefficient parallel to remote velocity, $\frac{X_p}{\rho n^2 D^4}$
D	aircraft aerodynamic drag parallel to remote velocity, pounds
D	rotor diameter, ft.

F_x, F_y, F_z	total aircraft forces along the X, Y, and Z axis respectively, pounds
$\overline{GJ}$	equivalent first mode blade torsional stiffness including control system flexibility, $lb\text{-}in^2$
I_x, I_y, I_z	aircraft moment of inertia about the roll, pitch, and yaw axis respectively, $slug\text{-}ft^2$
i_T	unit horizontal tail incidence, degree
J	propeller advance ratio, $\frac{V}{nD}$
L	aircraft aerodynamic lift normal to remote velocity, pounds
L_p	propeller force normal to remote velocity, pounds
L,M,N	total aircraft moments about the X, Y, and Z axis respectively, ft-lb
M	Mach number
MAC	mean aerodynamic chord, ft.
m	aircraft mass, slugs
N_I	primary gas generator RPM
$N_{II_{MAX}}$	Maximum allowable power RPM
$N_{II_{OPT}}$	Optimum power turbine RPM
N_{II}^*	Optimum power turbine RPM at static sea level Maximum power conditions
N_p	propeller force normal to shaft, pounds
n	propeller speed, rev/second
P	propeller power, ft-lb/second
q	dynamic pressure, $1/2\rho V^2$, lb/ft^2

R	propeller radius, D/2, ft
S	wing reference area, ft^2
T	turbine inlet temperature, degrees F
T	propeller thrust, pounds
u,v,w	perturbation velocities along the X, Y, and Z axis respectively, ft/sec
V	aircraft flight speed, knots or ft/sec
V_{Tip}	propeller tip speed, ft/sec
V_{STALL}	aircraft trim speed at $C_{L_{MAX}}$, knots, ft/sec
w_f	fuel flow, lb/hr
X,Y,Z	body axis coordinates, X positive forward, Y positive to starboard, and Z positive down from c.g.
X_p	propeller force parallel to remote velocity, pounds
α	aircraft angle of attack, degree
α_p	propeller shaft angle relative to fuselage reference, degree
β	aircraft sideslip angle, degree
$\beta_{.75}$	blade pitch angle at 75% radius, degree
δ	pressure ratio
δ_F	wing flap deflection angle, degree
η_c	cruise efficiency
ρ	air density, slugs/ft^2
ρ_o	air density at sea level, standard, day slugs/ft^2

σ	propeller solidity, $\frac{bc}{\pi R}$
Φ, Θ, Ψ	aircraft attitudes about the X, Y, and Z axis respectively
Θ	temperature ratio
Ω	propeller speed, rad/seconds
$(\dot{\ })$	derivative with respect to time
$\frac{d(\ )}{d(\ )}$	aircraft moment and force stability derivatives

SECTION I
INTRODUCTION

1. OBJECTIVE

The objective of Phase 1 work reported herein was to perform the preliminary design work necessary to establish a prop/rotor aircraft configuration that will meet the requirements of a specific transport mission. This configuration definition was necessary so that in Phase II a more detailed design of the prop/rotor, nacelle, wing and associated controls can be performed.

2. APPROACH

The Contractor's V/STOL Aircraft Sizing and Performance Computer Program (VASCOMP) was used to provide a matrix of designs meeting the basic mission with various payloads, disc loadings, and the associated gross weights. From these data a design was selected as a baseline configuration for further refinement. This involved trade offs of major items that were held constant in the initial sizing.

Additional studies were then made of aeroelastic stability, flying qualities and weight substantiation. The results were incorporated in a refinement of the design. The ability of this aircraft to perform a rescue and an alternate transport mission was then calculated.

SECTION II

SUMMARY

A preliminary design of a prop/rotor aircraft has been conducted and trade studies have been developed to show the impact on the gross weight of this aircraft which result from providing the major mission performance requirements of the selected transport mission. The aircraft designed appears to be practical and appears to be competitive with other configurations being considered for such a mission. The trade studies show that this aircraft is very sensitive to maneuver load factor requirements particularly in hover. Dash speed requirements increase the aircraft gross weight to accomplish the specified transport mission by 160 pounds per knot. The aircraft does not have unusual sensitivity to detail design assumptions except for the hovering disc loading.

A prop/rotor aircraft which can perform a transport mission with a 250 nautical miles radius, a cruise speed of 350 knots and a payload of five tons with a vertical take-off at 2,500 ft. and 93°F has been defined. This aircraft also can perform a rescue mission with a 500 nautical mile radius and a mid point hover time of thirty minutes. Landing gear sized to provide a coverage of 40 and 38 passes when operated on CBR4 soil is included in this design. A 21 percent wing thickness is used to provide the largest depth of wing compatible with high speed drag rise in order to satisfy the structural requirements of a prop/rotor aircraft with a minimum weight wing. The prop/rotor utilized has no rotor blade flap or lag hinges. Rotor blade cyclic pitch is planned to provide both control moments and load alleviation. A hover figure of merit of 75 percent and a cruise efficiency of 78 percent are expected to be achieved with this aircraft.

Weights substantiation of the preliminary design has been based on a conservative extrapolation of technology to the 1972 time frame. The weights methodology used is a mixture of airplane and helicopter trend curves. A 12.5 percent material factor has been applied to wing, tail and body group trend weights for 1972 materials. Engine section (nacelle, engine mount, etc.) group weight was reduced 9 percent for 1972 materials. Advanced gearing is showing such promise in present developments that a 15 percent factor on drive system weight has been taken for 1972 technology. No advances have been taken in the rotor group other than to assume that titanium hubs and fiberglass blades would be used. Similarly, there has been no advance taken in the weight of the flight control group. The weight benefit for advanced materials is more than offset for the wing by the requirements for vertical flight with the rotor thrust at the wing tip. The weight penalty for vertical flight was taken as a 25 percent addition to the wing trend weight. These assumptions are believed to be conservative but consistent with the unknowns of the use of hingeless rotors in a tilt rotor application.

The empty weight breakdown of the preliminary design aircraft is summarized as follows:

Group	Weight	Percentage of Design Gross Weight
Rotors	5,510	8.
Drive	7,282	11.
Flt. Controls	5,453	8.
Engines (Complete)	5,116	8.
Fuel System	1,652	2.
Wing	4,993	7.
Body	5,518	8.
Tail	1,266	2.
Landing Gear	2,571	4.
Elect. and Electronics	2,369	4.
Cargo Loading	981	1.
Other	3,150	5.
WEIGHT EMPTY	45,861 Lbs.	68
Fixed Useful Load	915	1.
Fuel	10,224	16
Payload	10,000	15
USEFUL LOAD	21,139 Lbs.	32
GROSS WEIGHT	67,000 Lbs.	100

The trade studies involved variations in mission parameters and aircraft design parameters. The results of these studies may be summarized in the following table:

Mission Parameter	Sensitivity
Dash Speed (At Speeds 300 Knots)	160 lb./knot
Maneuver Load Factor	
- Airplane	4,000 lb./g
-Hover	11,000 lb./g

Airplane Design Parameter	Sensitivity
Disc Loading	-2,300 lb./psf
Wing Loading (Chord Variation)	None-Optimized
Hover Tip Speed	-30 lb./fps.
Hover/Cruise RPM	None-Optimized
Tail Volume Coefficient	
-Vertical	33,300 lb. at 0.014
-Horizontal	6,000 lb. at 1.04
Parasite Drag	1,400 lb/ft^2
Afterbody Length	208 lb/ft
SFC	60,000 lb/lb-HP hr.

These sensitivities generally indicate that the aircraft is fairly well optimized. The disc loading sensitivity is negative since the disc loading of this aircraft is low compared to the dash speed requirement. This disc loading was selected to provide good helicopter mode operation and to provide for growth to a matched configuration.

SECTION III
CONFIGURATION DESCRIPTION

1. APPROACH TO STUDY

The purpose of the Phase I effort is to establish the aircraft general configuration so that in Phase II meaningful design work may be accomplished on the wing nacelle, rotor propeller and associated controls.

Therefore a broad look has been taken of the total aircraft so that the performance, weights, and geometric dimensions are realistic for the intended missions.

2. MISSION DEFINITION AND DESIGN CRITERIA

As the result of direction from the Flight Dynamics Laboratory and studies conducted by Boeing the following are the missions, design requirements and configuration decisions effective at the completion of Phase I for the Prop Rotor transport mission and for the rescue mission. The aircraft is sized to fly the transport mission. Its performance capabilities for the rescue mission are determined.

A. Transport Mission

For this mission the aircraft shall have a payload of 5 tons and have a cargo tie down system compatible with the 463L pallets. At overload gross weights, the cargo space shall be suitable for an 8-1/2 ton payload. A crew of 3 is used.

The design is not to be constrained by external noise or autorotative requirements.

The landing gear is to be compatible with a running take off, at overload gross weight (8½ ton payload), from a semi-prepared runway. It shall be designed for a sink speed of 12 ft/sec at normal gross weight.

The following are the segments of the mission:

1. Warm up and taxi; 2 minutes (MIL C-5011A),
2. Take off and hover, 1 minute, 2500' 93° (at MIL power or less)
3. Transfer to std temp at 2500' (no time and fuel allowance)
4. Climb at max R/C from 2500' to 10,000 feet. Standard day at MRP
5. Cruise at 350 knots to 150 N. Miles from base at 10,000' standard. (400 knots dash capability must be available at MIL power.)
6. Descend to sea level (no time and fuel allowance)
7. Dash at 300 knots for 100 nautical miles at sea level standard at MIL power or less.
8. Transfer altitude to 2500' 93° (no time and fuel allowance)
9. Hover 2 minutes at 2500' 93°
10. Land and exchange payload (one minute time, no fuel allowance)
11. Warm up and taxi 2 minutes, (MIL C-5011A)

12. Take off and hover, 1 minute at 2500' 93°

13. Transfer to sea level standard (no time and fuel allowance)

14. Dash at 300 kt for 100 n. miles

15. Climb to 10,000 feet at max R/C at NRP on standard day

16. Cruise at 350 knots back to base on standard day

17. Descend to 2500' 93° (no time and fuel allowance)

18. Hover 2500' 93° for 2 minutes (no fuel allowance)

19. Land with 10% fuel left

NOTE: All SFC's to be increased.
Figure III-1 depicts this mission.

B. Rescue Mission

To perform the rescue mission the basic transport aircraft is modified. This involves: 1) the removal of the 463L cargo system and troop seats, 2) the addition of litters and seats, two machine guns, armor plate, medical equipment, two medics to the crew, rescue hoist, additional fuel and fuel tankage and 3) the changing of the electronic equipment to that required for rescue operations. A detailed listing of these changes is provided in the weights section of this report.

With the aircraft chosen for the transport missions, the cargo and useful load for the transport mission will be replaced by equipment and additional fuel tankage required for the rescue mission (no "snatch" system will be required.)

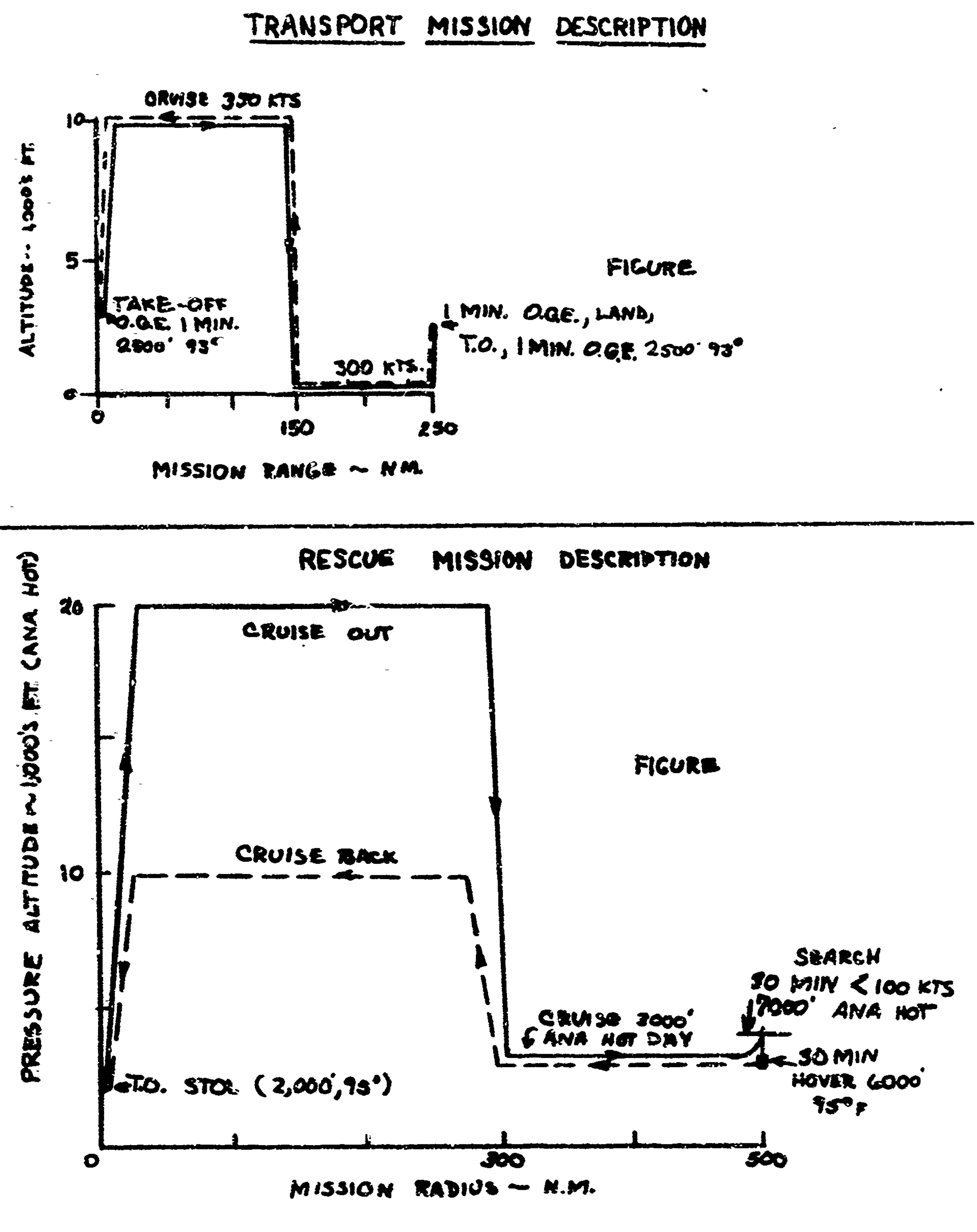


FIGURE III-1 TRANSPORT AND RESCUE MISSION DEFINITIONS

The following are the segments of the mission.

1. Warm up and taxi 2 minutes at NRP.
2. Take off 1 minute at Max Rated at 2000' 95° STOL
3. Climb at best R/C at NRP, ANA hot day
4. Cruise at NRP to 300 miles from base to 20,000 feet ANA hot day at NRP
5. Descend to 3000', ANA hot day (no time and fuel allowance)
6. Cruise at 350 knots at 3000', ANA hot day for 200 nautical miles
7. Climb at best R/C to 7000', ANA hot day at NRP
8. 30 minute search at 100 knots or less, 7000' ANA hot day
9. Descend to 6000' 95° (no time and fuel allowance)
10. 30 minutes hover at 6000' 95° at MIL. Pick up 1200 lb midway through hover.
11. Descend to 3000' ANA hot day (no time and fuel allowance)
12. Cruise at 350 knots at 3000' ANA hot day for 200 nautical miles
13. Climb at best R/C to 10,000' ANA hot day at NRP (This altitude is based on carrying injured persons without pressurization)
14. Cruise back to base at NRP, ANA hot day
15. Descend to 2000' 93° (no time and fuel allowance)
16. Land with 10% reserve fuel

NOTE: SFC's are to be increased 5% above specification values per MIL C-5011A

Figure III-1 depicts this mission.

C. Alternate Transport Mission

After a point design has been chosen, the following simple mission will establish the importance of emphasing the hover and forward flight mode. All SFC are increased by 5% in accordance with MIL-C-5011A.

1. Load aircraft with payload (P)
2. Warm up and taxi, 2 minutes (MIL-C-5011A)
3. Take off and hover one minute at 2500' 93°
4. Transfer to std. temp. at 2500'
5. Climb at max R/C from 2500' to 10,000' standard day, NRP
6. Cruise at normal rated power to radius (R)
7. Transfer altitude to 2500' 93° (no time and fuel allowance)
8. Hover for (H) minutes at 2500' 93°
9. Land and exchange payload (no time and fuel allowance)
10. Warm up and taxi 2 minutes at 60% mil power 2500' 93°
11. Take off and hover one minute at 2500' 93°
12. Transfer to 2500' standard day
13. Climb to 10,000' at max R/C at NRP standard day
14. Cruise at normal rated power back to base.
15. Descend to 2500' 93° (no time and fuel allowance)
16. Hover at 2500' 93° for 2 minutes (no time and fuel allowance)
17. Land with 10% fuel left.

Payload (P), Hover time (H) and Radius (R) are to be varied within fuel and payload available.

3. BASELINE CONFIGURATION

A. 3-View

Figure III-2 drawing SK215-21583 shows the general arrangement. The selection of a high wing configuration is to provide nacelle to ground clearances. The location of the engines and the complete drive (transmission) system within the tilting nacelles is based on reducing the dependence on the interconnecting shafting. The arrangement shown requires the use of crossshafting only to provide power between nacelles in the event of an engine out condition. An engine or an engine to nacelle shaft failure has the effect of a power reduction only. The loss of the cross shafting between nacelles has no effect on the aircraft if all engines are operating.

The wing is positioned on the fuselage and the nacelle tilt axis is located so that during normal c.g. the rotor thrust in the hover mode is through the c.g. In cruise flight the c.g. is at the 25% chord. This minimizes the amount of trim required and the change in trim during conversion.

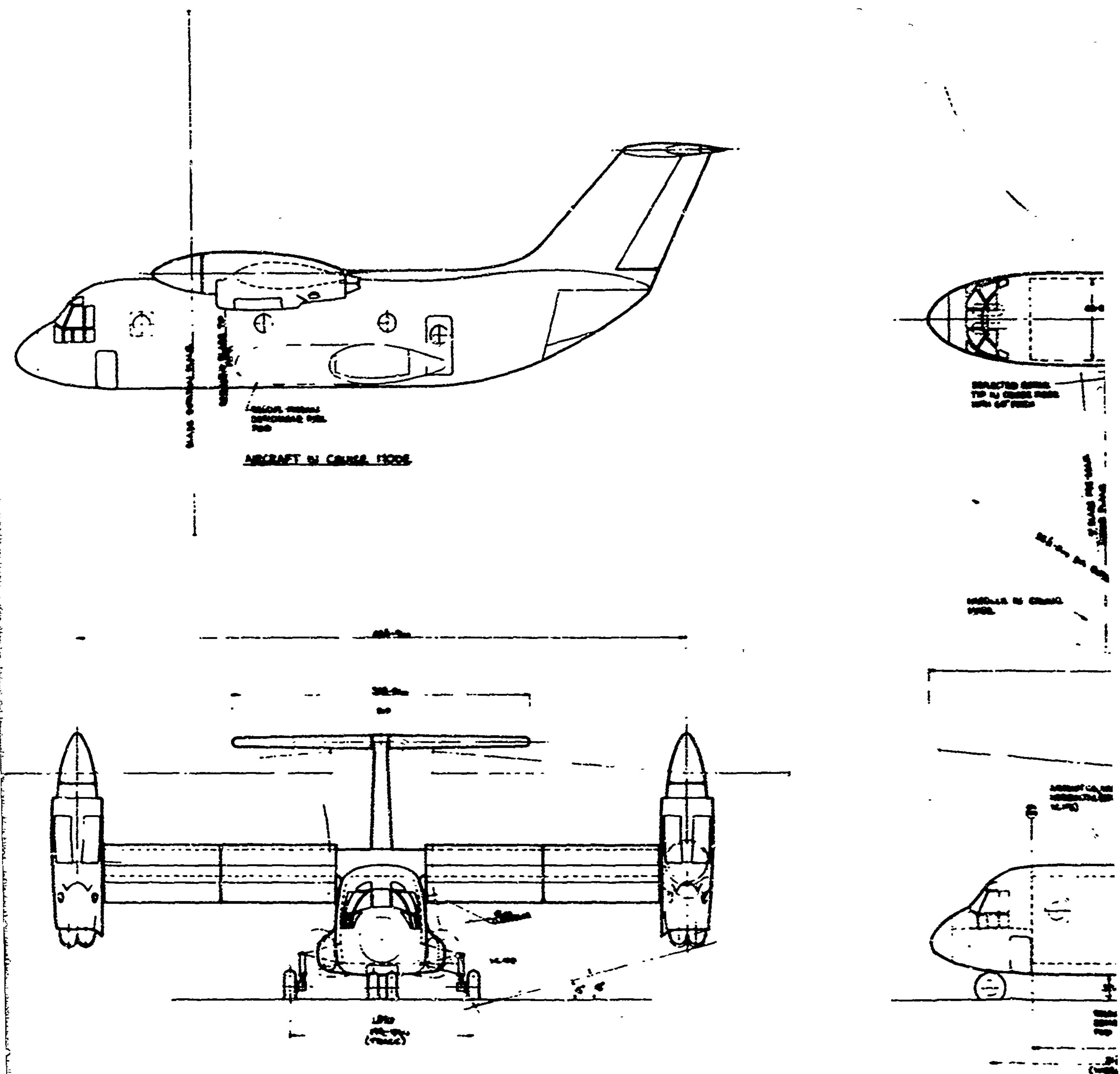
The clearance between the prop pitch axis and the wing leading edge has been selected at 4.5 ft for the USAF tilt rotor aircraft. This clearance is based on a structural design limit gust of 50 ft/sec at 400 knots (dash speed) 10,000 ft standard day.

This clearance enables the pivot to be positioned approximately at the shear center of the wing (38% $\bar{c}$), and with the thrust line in hover through the hover cg. The cruise cg is at 25% $\bar{c}$.

This clearance is considered to be conservative; however, better precision would require the use of power spectral models and probability assessment of gust occurrence and failures.

The cargo compartment size is 432 inches long, 104 inches wide and 110 inches high giving a volume of 2,874 cubic feet. This size has been used in other Air Force light transport studies.

The major components are discussed in the following paragraphs.



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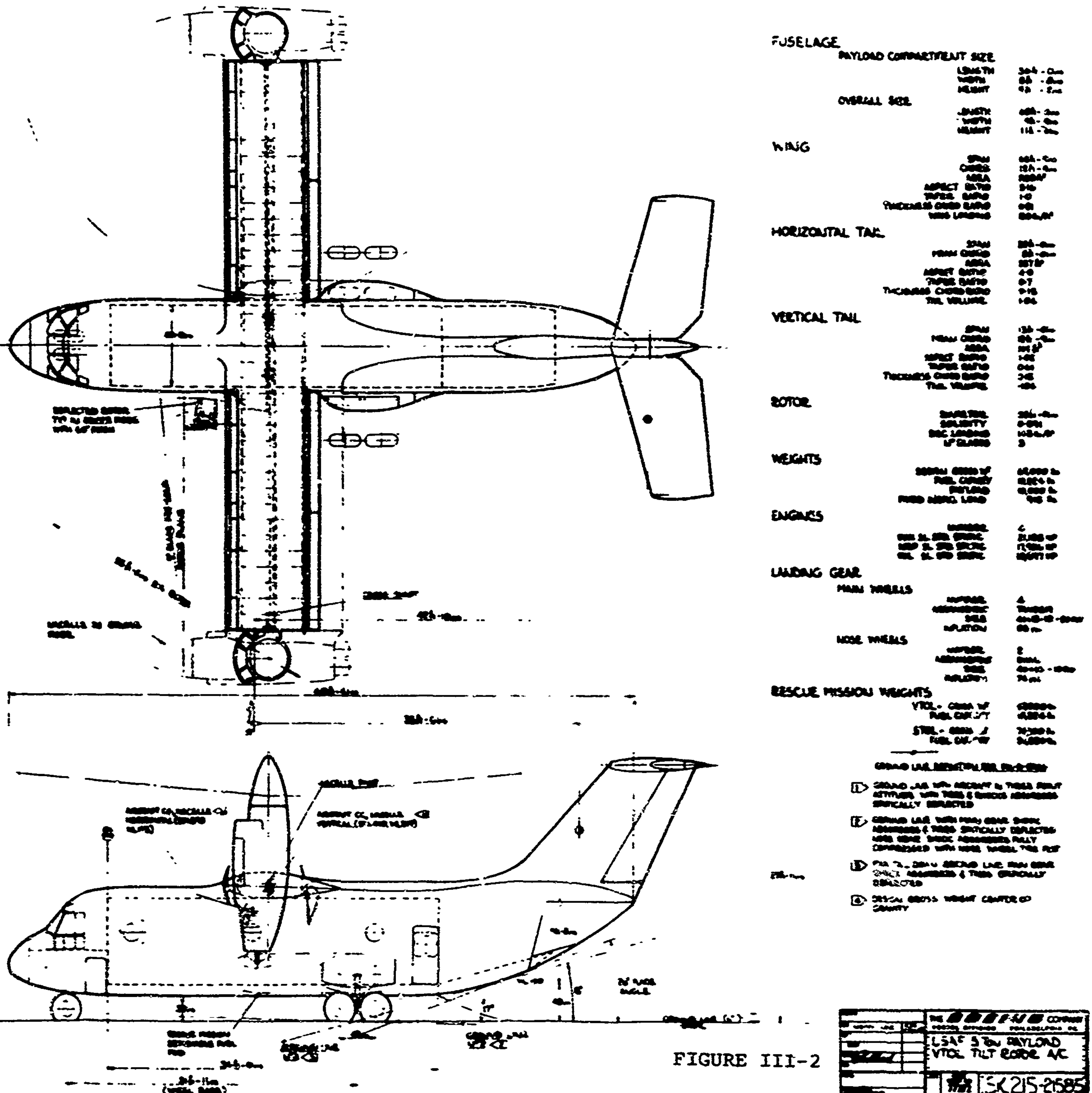


FIGURE III-2

Sheet III-9

B. Fuselage and Landing Gear

The fuselage was sized by the cargo compartment dimensions, crew compartment and loading ramp arrangement. The fairing of the aft end of the fuselage was studied to determine the minimum length that could be faired around a ramp and door arrangement which allowed loading of pallets and vehicles. This was a length 68'4" for the fuselage. Afterbodies which were longer (greater fineness ratios) were also studied for their effect on drag, tail arm and size and the resultant effect on gross weight. The results are shown on Figure III-13. The shortest length satisfied the loading condition and this produces a minimum gross weight.

The landing gear is of the tricycle arrangement with dual nose wheel and tandem wheels on each main gear. This arrangement was selected to provide low weight, small retracted gear volume and provide the coverage and passes satisfactory for the intended use of the aircraft.

The sizing of a landing gear for this machine was based on the following data, using Reference III-1 and -2.

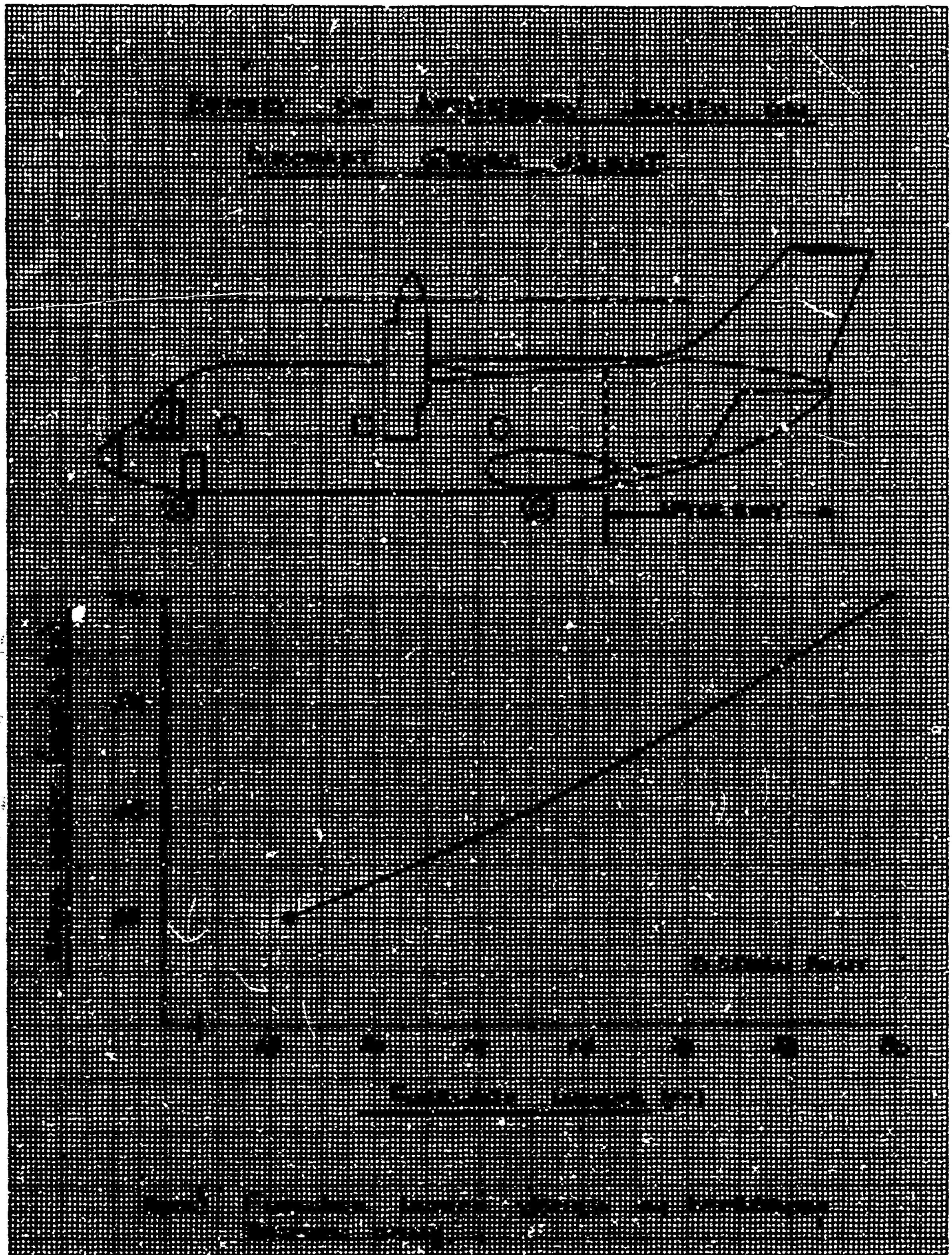


Fig. III-13 VARIATION OF MODEL 215 GROSS WEIGHT WITH FUSELAGE AFTERBODY LENGTH.

ANALYSIS OF USAF TILT ROTOR AIRCRAFT (SK215-21585) IN VERTICAL MODE

GW 74,000 lb
Nose Tires 40 x 14

Main Tires 41 x 15

NOSE WHEEL	MAIN WHEEL
1. Single Wheel Load (SWL)	1.
$SWL_N = \frac{GW\ (F-L)^*}{F \times N_N}$	$SWL_M = \frac{74000 \times (383-99.5)}{383 \times 4}$
$SWL_N = \frac{74000\ (383-256.3)}{383 \times 2}$	$SWL_M = \underline{13,693\ lb}$
$SWL_N = \underline{12,239\ lb}$	
2. Single Tire Contact Area (A)	2.
Deflection (dn) $= \frac{b(D_O-D_F)}{200}$	$= \frac{50}{200} \times (41.-21.25)$
$= \frac{50(39.8-19.25)}{200}$	
dn $= \underline{5.137"}$	dm $= \underline{4.94"}$
$A_N = 2.36\ d \sqrt{(D_O-d)(w-d)}$	$A_M = \underline{222\ sq\ in.}$
$= 2.36 \times 5.14 \sqrt{(39.8-5.14)(14-5.14)}$	
$A_N = \underline{212.5\ sq\ in.}$	
3. Contact Pressure (CP)	3.
$CP_N = SWL_N / A_N$	
$= \frac{12239}{212.5}$	$= \frac{13693}{222}$
$CP_N = \underline{57.59\ psi}$	$CP_M = \underline{61.68\ psi}$

*Symbols are defined in Figure III-3.

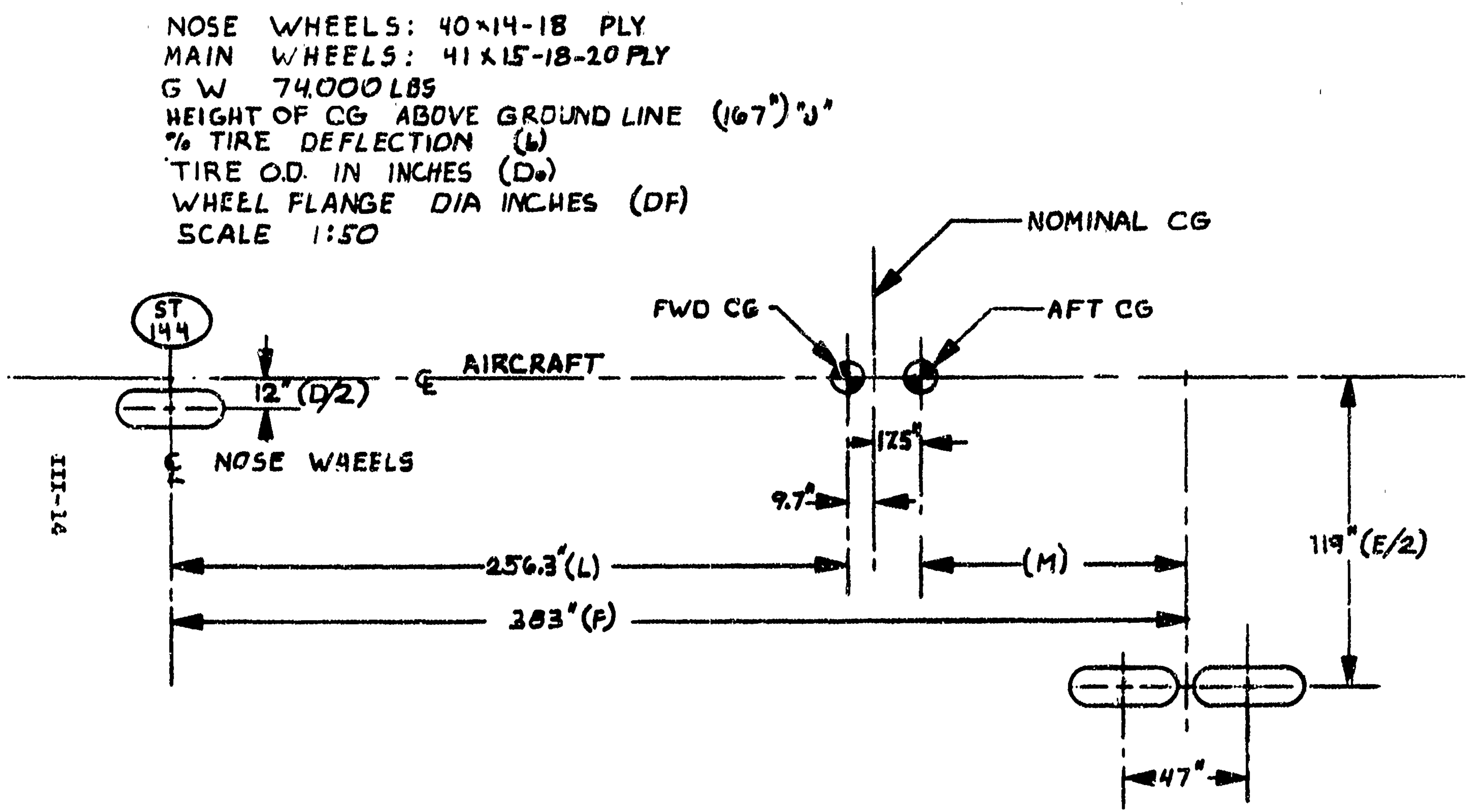


FIGURE III-3 . MODEL 215 LANDING GEAR ARRANGEMENT

NOSE WHEEL	MAIN WHEEL
4. Contact Area Radius (R)	4.
$R_N = \sqrt{212.5/\pi}$	$= \sqrt{222/\pi}$
$R_N = 8.24$	$R_M = 8.4"$
$B/R_N = 24/8.24$ $= 2.92$ radii	$D/R_N = 47/8.4$ $= 5.60$ radii
$ESWL_N = SWL_N + FACTOR \times SWL_N$	0% ∴
$= 12{,}239 + .58 \times 12{,}239$	$= 13{,}693$
$= 19{,}459$	
5. Coverages	5.
CBR = 4 $CBR_1 = 2.16$	CBR = 4 $CBR_1 = 2.12$
$C_N = \left(\frac{CBR}{CBR_1}\right)^6 = 40.0$	$C_M = 45.0$

ANALYSIS OF USAF TILT ROTOR AIRCRAFT (SK215-21585) IN STOL MODE

Max Deceleration Rate 6ft/sec^2 ($\ddot{\sigma}$)
GW 74,000 lb
Nose Tires 40 x 14 Main Tires 41 x 15

<table>
<tr><th>NOSE WHEEL</th><th>MAIN WHEEL</th></tr>
<tr><td>1. Single Wheel Load (SWL)
$SWL_N = \frac{GW\ (F-L)}{FxN_N} + \frac{\ddot{\sigma}\ xGWxJ}{32.2xFxN_N}$
$= \frac{74000(383-256.3)}{383 \times 2} + \frac{6x74000x167}{322 \times 383 \times 2}$
SWL_N = 15,346 lb</td><td rowspan="4">MAIN WHEEL ANALYSIS FOR STOL MODE AS VERTICAL MODE UP TO COVERAGES CALCULATION</td></tr>
<tr><td>2. Single Tire Contact Area (A)
From Sh. 1 d_N = 5.14"
A_N = 212.5 sq in.</td></tr>
<tr><td>3. Contact Pressure (CP)
$CP_N = SWL_N/A_N$
$= \frac{15346}{212.5}$
CP_N = 71.5 psi</td></tr>
<tr><td>4. Contact Area Radius (R)
From Sh. 1 R_N = 8.24
B/R_N = 2.92 radii
$ESWL_N = SWL_N + \text{Factor} \times SWL_N$
= 15,346 (1+.58)
$ESWL_N$ = 24,250</td></tr>
</table>

NOSE WHEEL	MAIN WHEEL
5. Coverages	
CBR = 4 CBR_1 = 2.95	
$C_N = \left(\frac{CBR}{CBR_1}\right)^6 = 6.2$	
6. Passes/Coverage Ratio	**6.**
$P/C_N = \frac{D+80+W_N}{.75 x N_N x W_N}$	
$= \frac{24+80+12.7}{.75 \times 2 \times 12.7}$	$= \frac{47+80+13}{.75x4x13}$
$P/C_N = 6.13$	$P/C_M = 3.59$
7. Passes Calculations (P)	**7.**
$P_N = C_N \times P/C_N$	= 40. x 3.59
= 6.2 x 6.13	$P_M = 143.8$
$P_N = 38.0$	
Aircraft Passes	
$AP_N = \frac{80x38x143.8}{80x143.8+(80-80)38+(80-80)38}$	
$AP_N = 38$	$AP_M = 144$

Gear Design Gross Wt. (lb)	74,000
Take-Off and Landing Mode	Vertical
C.B.R. (%)	4
Coverages	40
Aircraft Passes (6 ft/sec decel.)	38
CG Limits Stn. (ins)	$410^{-9.7}_{+17.5}$
W.L. (ins)	219
B.L. (ins)	℄ Aircraft
Sink Speed (ft/sec)	12
Total Vertical Travel of Gear (ins)	13.4
Limit Load Factor at Gear (g)	2
Allowable Tire Deflection (%)	50
Overturning Angle (deg)	27

A landing gear designed to the above criteria was analyzed for STOL landings at a deceleration rate of 6 ft/sec^2 to determine the number of "aircraft passes". This was calculated to be 38. The analyses are presented on the following pages.

The wheel and tire selections are based on the comparisons shown in Figure III-4 and Figure III-5. Figure III-4 is for a wheelbase of 383 inches. This is used on Model 215.

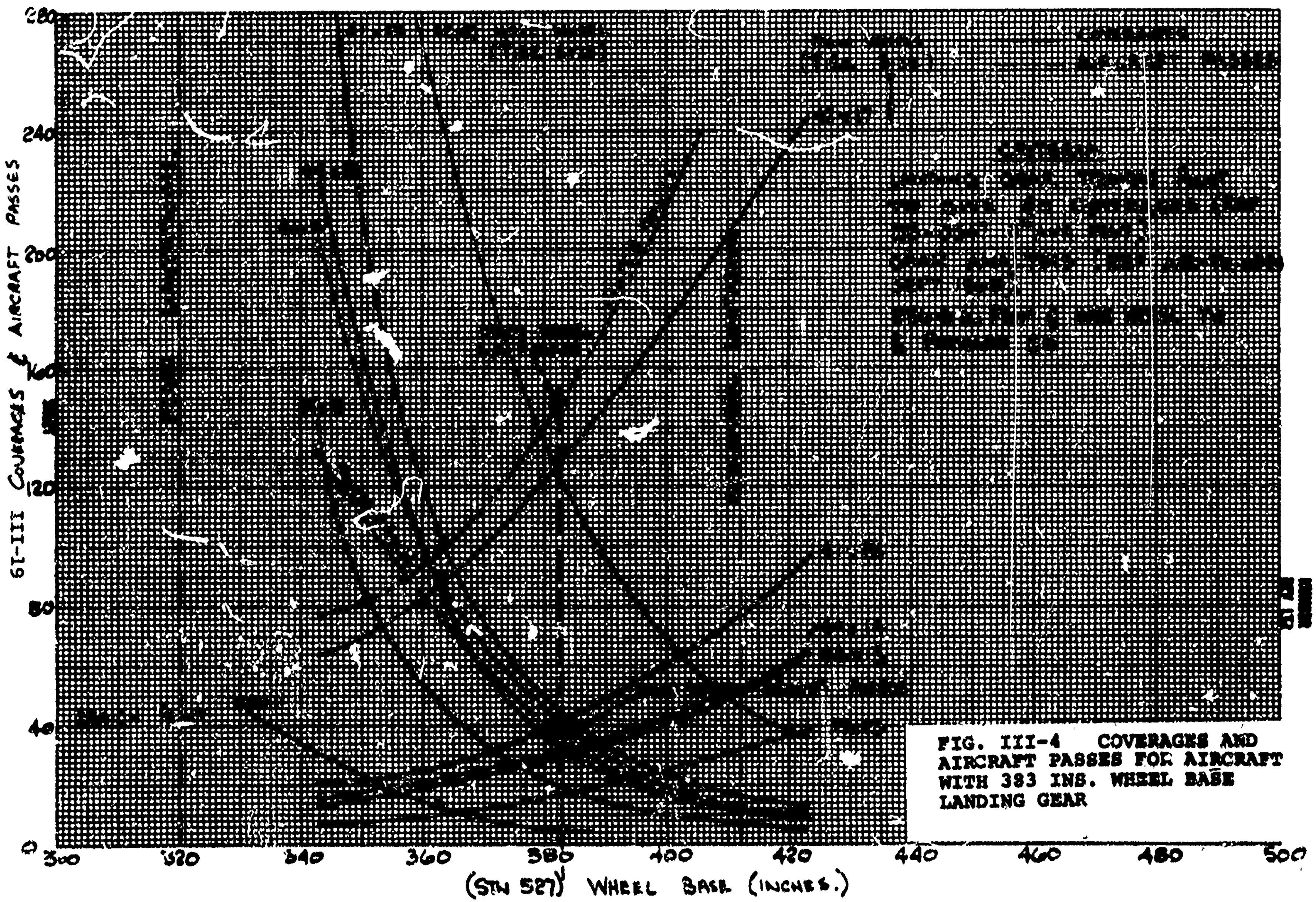


FIG. III-4 COVERAGES AND AIRCRAFT PASSES FOR AIRCRAFT WITH 383 INS. WHEEL BASE LANDING GEAR

Figure III-5 is for a 432" wheelbase to show the effect of wheelbase on wheel and tire sizes as a function of number of passes and coverage.

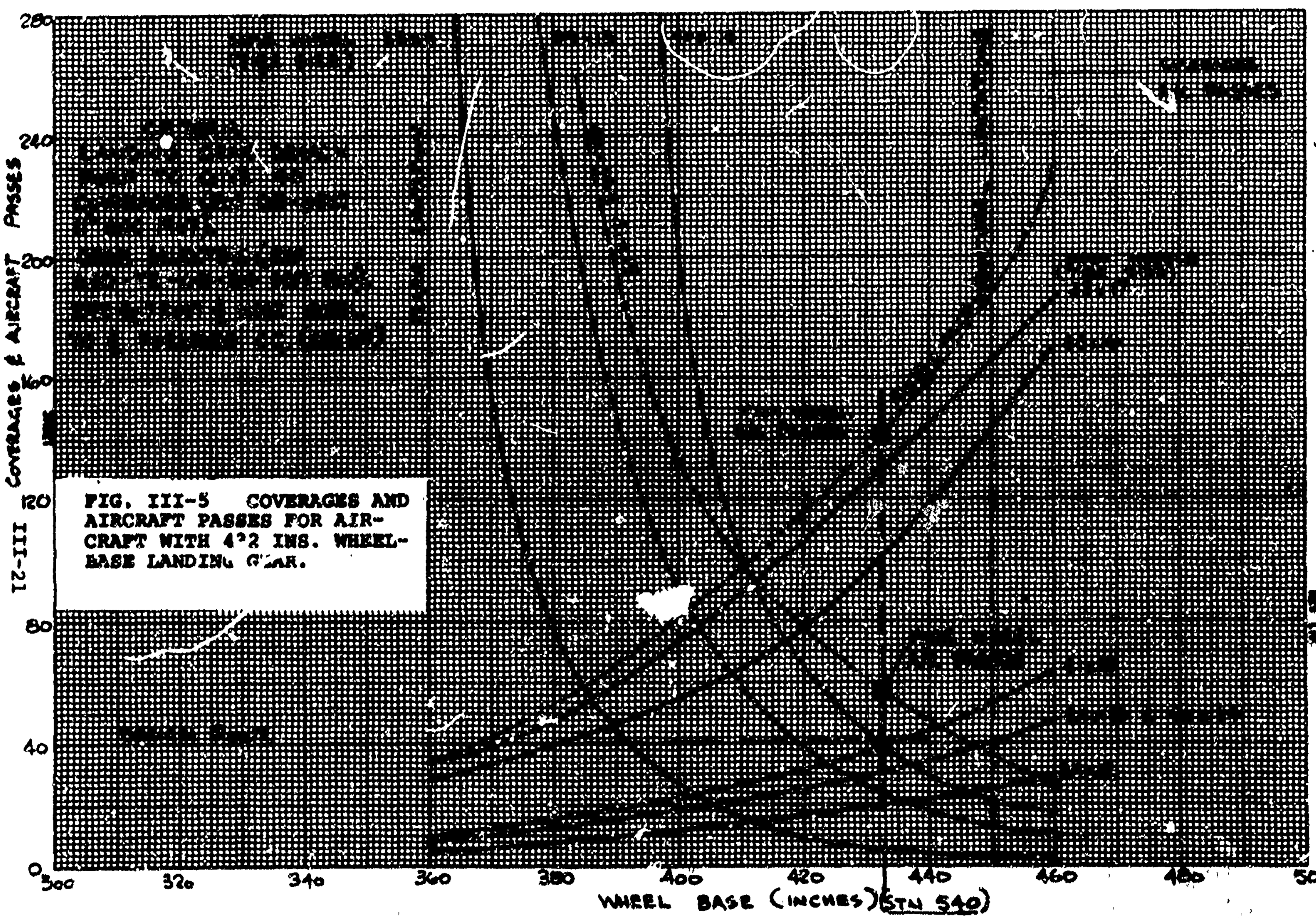


FIG. III-5 COVERAGES AND AIRCRAFT PASSES FOR AIRCRAFT WITH 4?2 INS. WHEEL-BASE LANDING GEAR.

C. EMPENNAGE

A unit horizontal tail was selected since it is lighter than a stabilizer and elevator. A powered control will be required in either case.

The "T" configuration was chosen to reduce the size of the vertical fin. The fin weight per surface area is greater but a small weight advantage is anticipated.

The tail sizing is discussed under the flying qualities section.

D. WING

The wing thickness is 21% to minimize wing weight. The drag divergence Mach No. for this section at 10,000 ft standard day has been calculated at $M_{DD} = 0.618$. The flight Mach No. at 400 knots is 0.626. At 400 knots we are exceeding M_{DD} by $\Delta M = 0.008$.

Trailing edge separation could occur; triggered by compressibility and by large trailing edge angles. This phenomenon would of course reduce the effectiveness of aileron controls significantly.

This problem can be alleviated by tailoring the position of maximum thickness on the wing and also by the use of vortex generators.

Phase II of the contract is aimed at a detailed wing and rotor design and as such is the proper time to assess this problem in greater depth.

The wing chord has been kept constant and no wing sweep is used. Only a small amount of forward sweep is attainable without complicating the cross shaft and the fuel tank installation.

Taper was avoided based on previous work which showed that the nacelle pivot structure could be made lighter with the greater depth achieved with an untapered planform. This saving was greater than the weight penalty at the wing root of the untapered wing. This decision will be reviewed as part of Phase II wing design.

Fuel is carried in crash resistant self-sealing tanks in the wing. The tanks would also be provided with a flame suppression system.

The leading edge of the wing is fitted with a download reduction device which is also intended to prevent skittishness when hovering close to the ground. Simple trailing edge flaps are fitted which when fully deflected also serve this purpose. They are used as flaps and ailerons during late transition and as ailerons in cruise flight.

E. Nacelle Arrangement

Drawing SK215-21584, Figure III-6, shows the nacelle arrangement. A large tube is fixed to each wing tip. Two bearings are located on this tube to attach the nacelle. The cross shafting, mechanical controls, fluid lines and wireing pass from the wing to the nacelle through this tube so that they are located at or near the center of rotation.

A truss structure attaches to the bearings at the pivot and supports the transmission at the other end. The transmission is mounted at the forward end of the truss so that the large moments and forces from the prop/rotor do not go through the transmission case. This prevents case deflections which would distort the transmission ring gears and bearing.

The engines are mounted in the opposite end of the nacelle. The inlet duct has a high ram recovery configuration which would be used in the airplane mode. An alternate air door and filter system is fitted for use in the hover and low speed flight regime. This prevents sand and dirt ingestion during hover and STOL operation which can cause rapid engine deterioration.

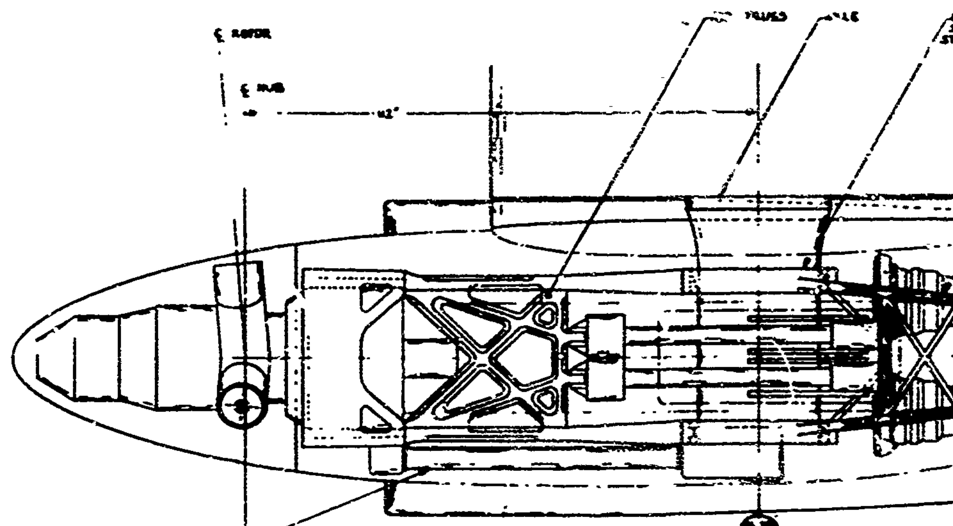
Accessories and coolers, are also shown. An exhaust deflector is fitted to the engines to minimize the possibility of setting fires when operating from fields with dry grass. These deflectors retract in airplane flight and serve to size the tail pipe to the desired area for cruise flight.

F. Engine Selection

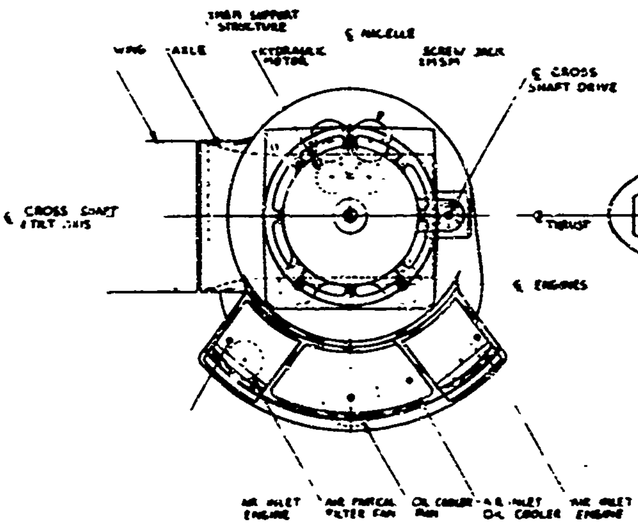
In the power class required for this aircraft there are no free turbine engines of current technology in production. However, General Electric, Allison and Lycoming all have developments of engines from which the required engine could be based. The Lycoming LTC4V1 free turbine engine was chosen since it has run as a complete engine at the required power and demonstrated specific fuel consumption that is as good as that used in this study. The other engines, when scaled to the required power, would have very similar characteristics. The selection of a specific engine was not a major factor in the overall aircraft preliminary design.

G. Prop Rotor

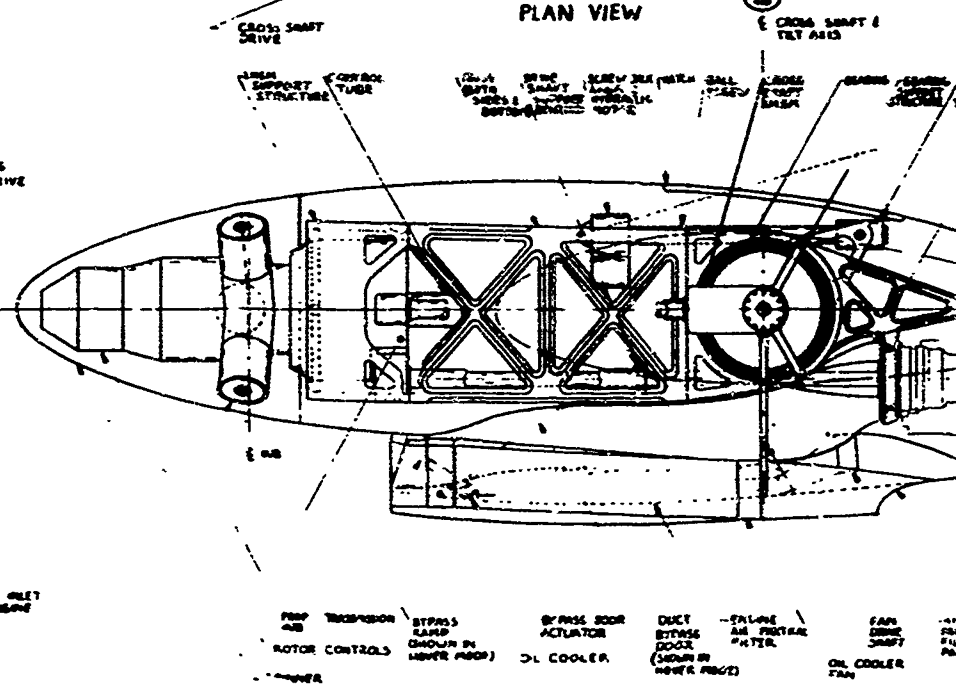
The prop rotor which will be designed in Phase II is a hingeless or "rigid" rotor. It has a hub which provides pitch change, both cyclic and collective. There are no



PLAN VIEW



VIEW LOOKING AFT



ELEVATION VIEW LOOKING INBRD

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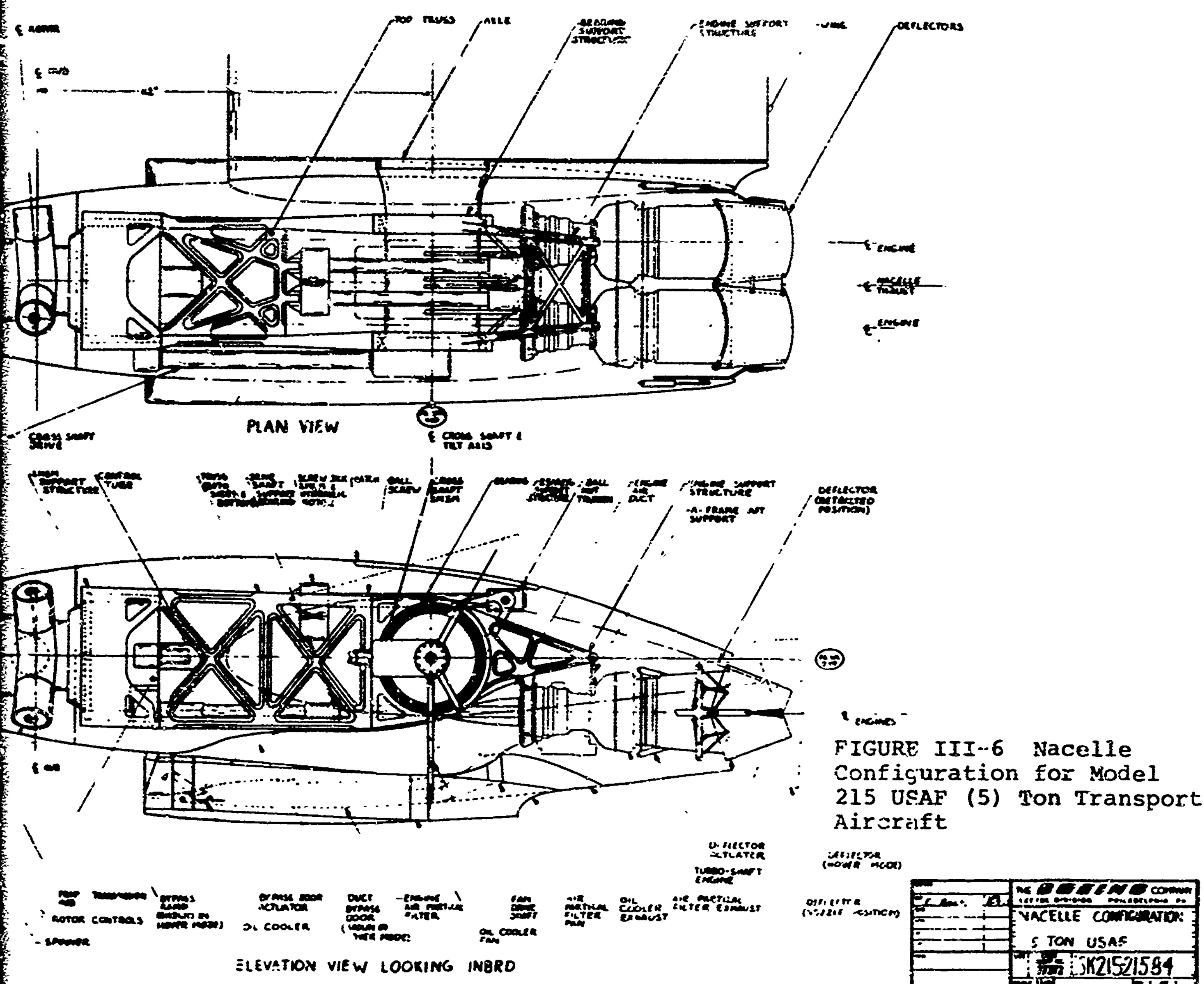
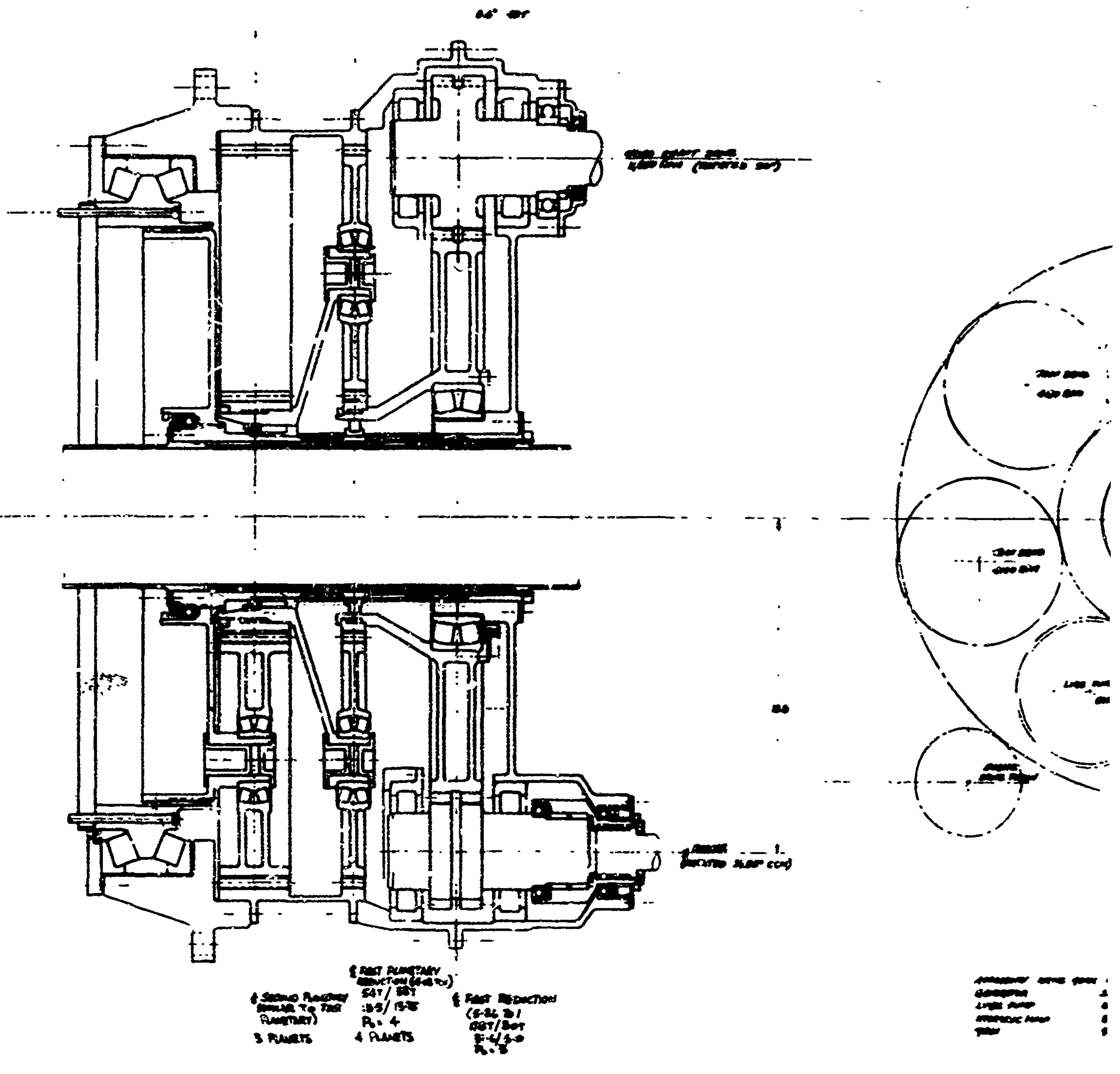


FIGURE III-6 Nacelle Configuration for Model 215 USAF (5) Ton Transport Aircraft



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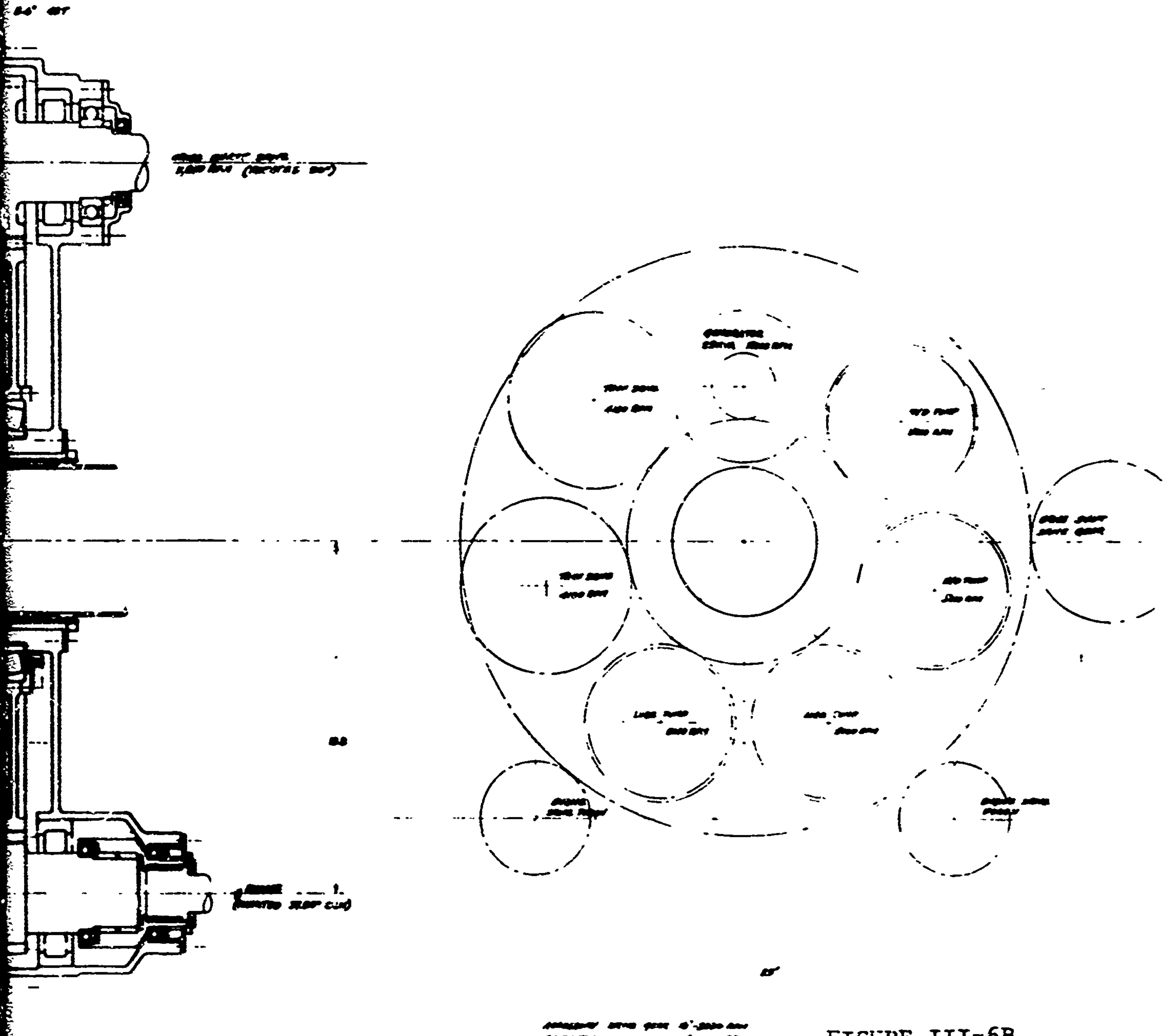

FIGURE III-6B

THE BOEING COMPANY
VERTOL DIVISION PHILADELPHIA PA
USAF 5 TON PAYLOAD
VTOL TILT ROTOR
TRANSMISSION
SK215-21586

Sheet III-29

B

mechanical hinges for flap or lag motion.

The first coupled mode frequency (chordwise) at hover rotational speed will be approximately .75 per rev and the second coupled mode (flapwise) will be approximately 1.2. These frequencies are selected to permit design of a blade with relatively low blade root moments and stresses. Also the first mode frequency is high enough to avoid mechanical instability with a small amount of damping.

The blades are made of composite materials permitting design freedom to readily vary the blade stiffness and shape. The cyclic and collective controls are contained within the rotor propeller housing so that lubrication is provided with a minimum number of seals and sand is excluded.

4. SENSITIVITY AND TRADE STUDIES

Figure III-7 shows an overall study of prop/rotor diameter, VTOL payload and gross weight. It can be seen that for a given payload, the disc loading effect on gross weight is relatively small in the range from 14 to 20 pounds per square foot. Experience with low disc loading VTOL aircraft (helicopters) has shown that in their production life the gross weight increases about 50% as a result of additional equipment and increased payload capabilities. For this reason the design was chosen to have a disc loading of 14. This would mean that a disc loading of approximately 21 would be achieved in the later production versions.

The impact of the airplane flight load factor on weight is shown in Figure III-8. The sensitivity is shown to be about 2000 lb/g.

A curve of hover tip speed and gross weight is shown in Figure III-9. The major saving in weight with high tip speeds is in the transmissions because of the lower torque and in the rotor because of a reduction in blade area. The compressibility losses associated with high rotor tip speeds in cruise (i.e. Mach Number effects) as well as noise are deterrents to the higher tip speeds.

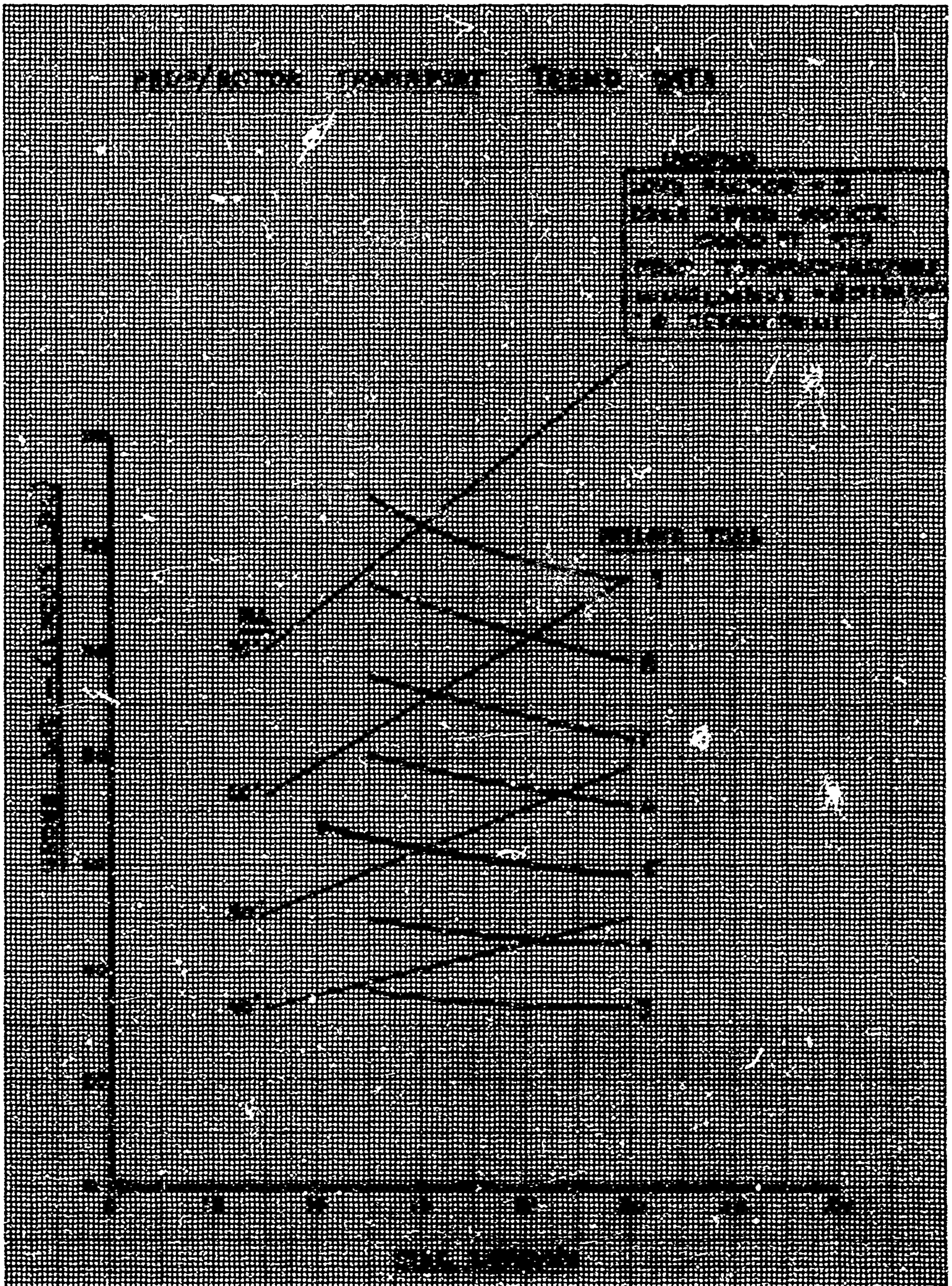

FIGURE III-7 VARIATION OF MODEL 215 GROSS WEIGHT WITH PAYLOAD AND DISC LOADING

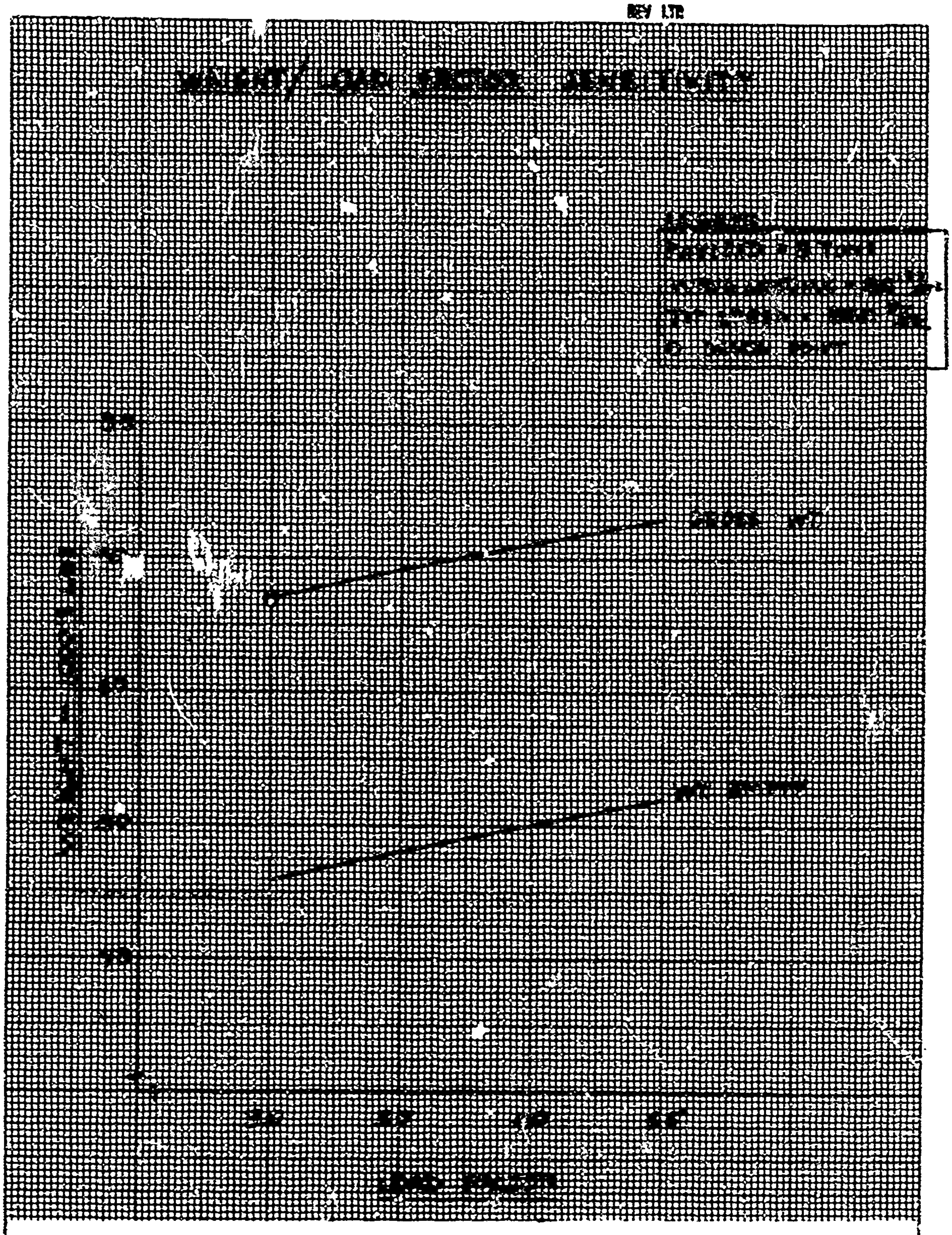

FIGURE III-8 VARIATION OF MODEL 215 GROSS WEIGHT WITH THE STRUCTURAL LIMIT LOAD FACTOR IN AIRPLANE MODE.

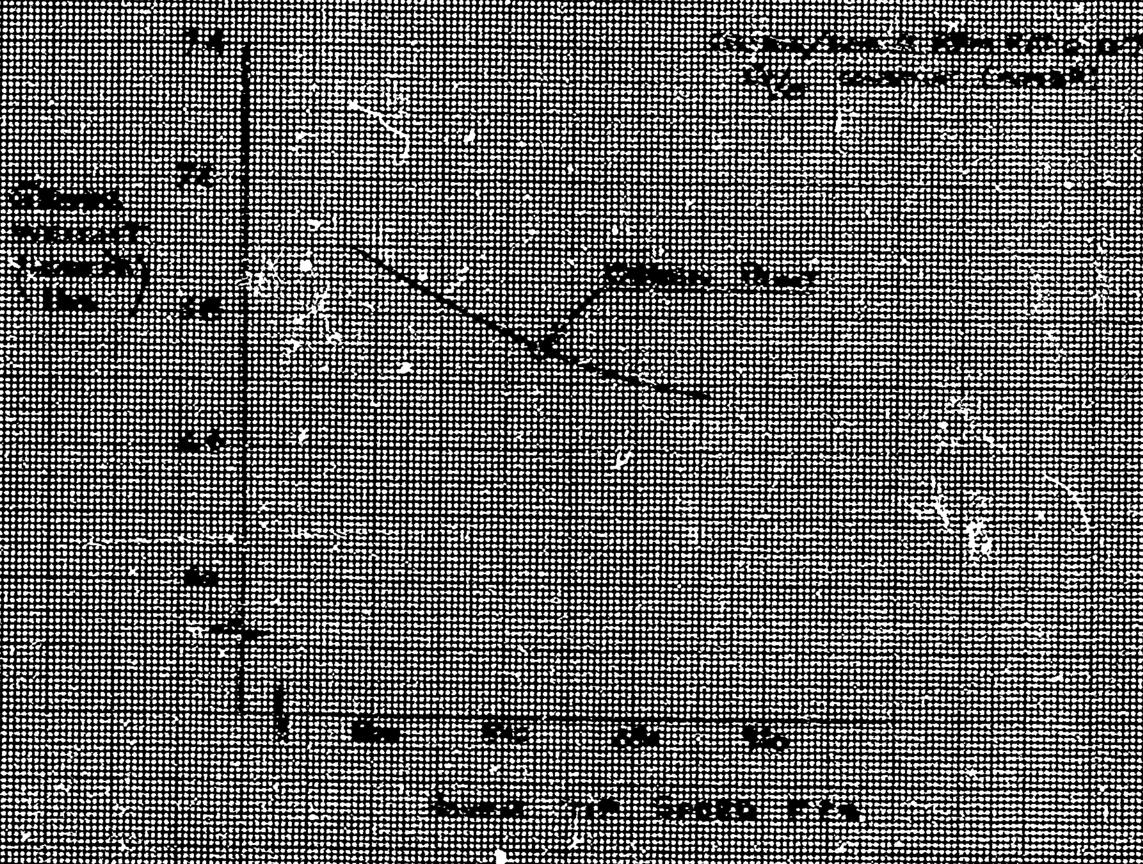

FIGURE III-9 VARIATION OF MODEL 215 GROSS WEIGHT WITH HOVER TIP SPEED

Figure III-10 shows the gross weight change as a function of speed. Since the aircraft has a relatively low disc loading the power required in hover is low while the requirement to overcome airplane drag at high speed sizes the engines and transmission. The power reduction with reduced speed is reflected by the gross weight reduction.

The effect of the percent reduction in rpm from hover to cruise is shown in Figure III-11. The reduction of prop/rotor rpm at cruise is desireable in order to provide higher blade loading. However the reduced rpm results in higher transmission torque and therefore weight. Moreover if the speed change is to be accomplished without the complexity of a gear shift, the design of the power turbine and the fuel control system must be considered. The power turbine with its hover rpm 5% above optimum and with a 30% reduction in rpm for cruise provides the greatest realizable weight saving. Larger speed reductions result in increased specific fuel consumption and weight negating the improved prop/rotor efficiency.

The Figure III-12 shows the effect of tail area on gross weight. The tail volume chosen is dictated by stability and flying qualities and is discussed in Section 4.

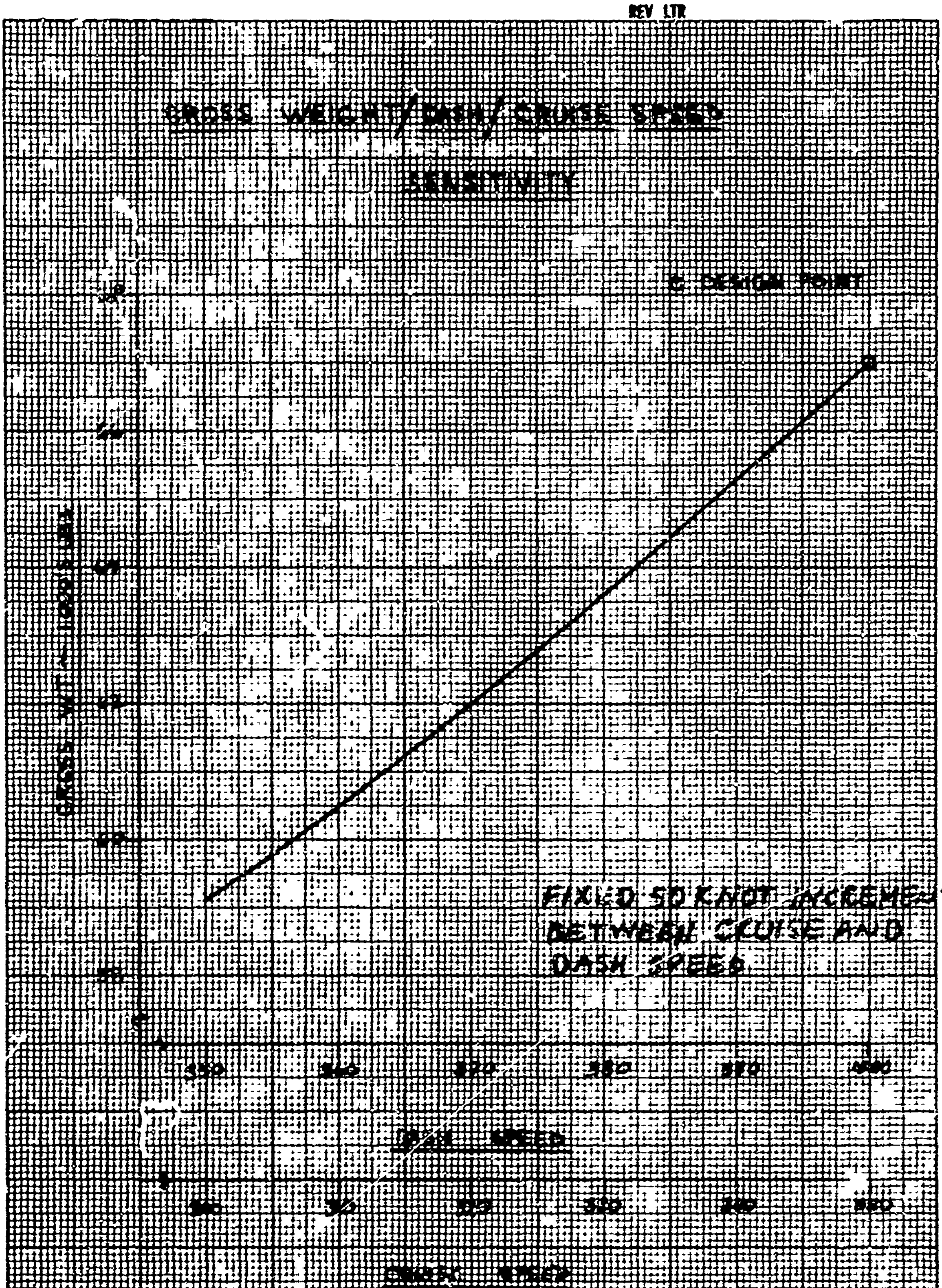

FIGURE III-10 VARIATION OF MODEL 215 GROSS WEIGHT WITH CRUISE AND DASH SPEED REQUIREMENTS

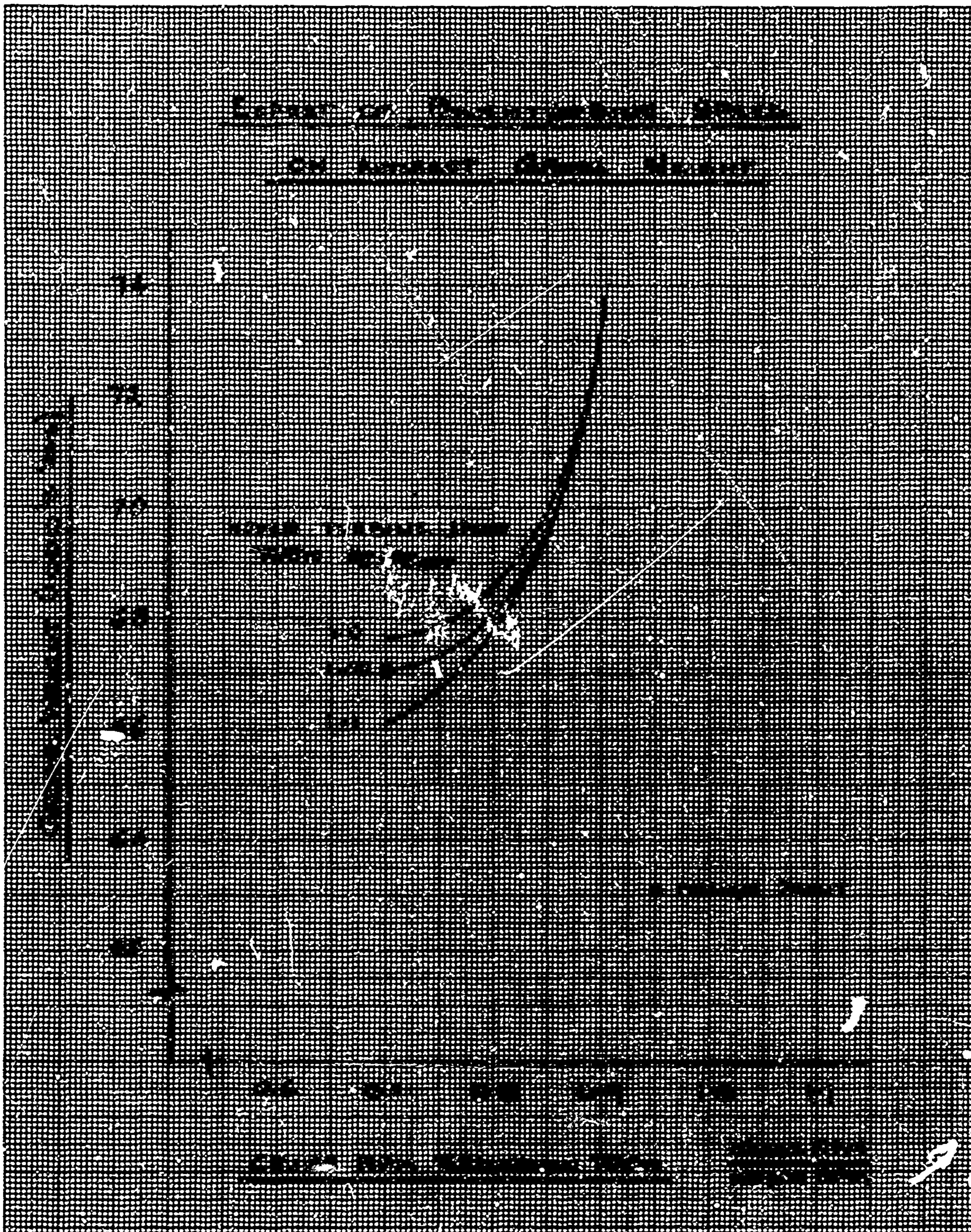

FIGURE III-31 VARIATION OF MODEL 215 GROSS WEIGHT WITH THE REDUCTION OF CRUISE R.P.M. AND WITH POWER TURBINE OVERSPEED IN HOVER.

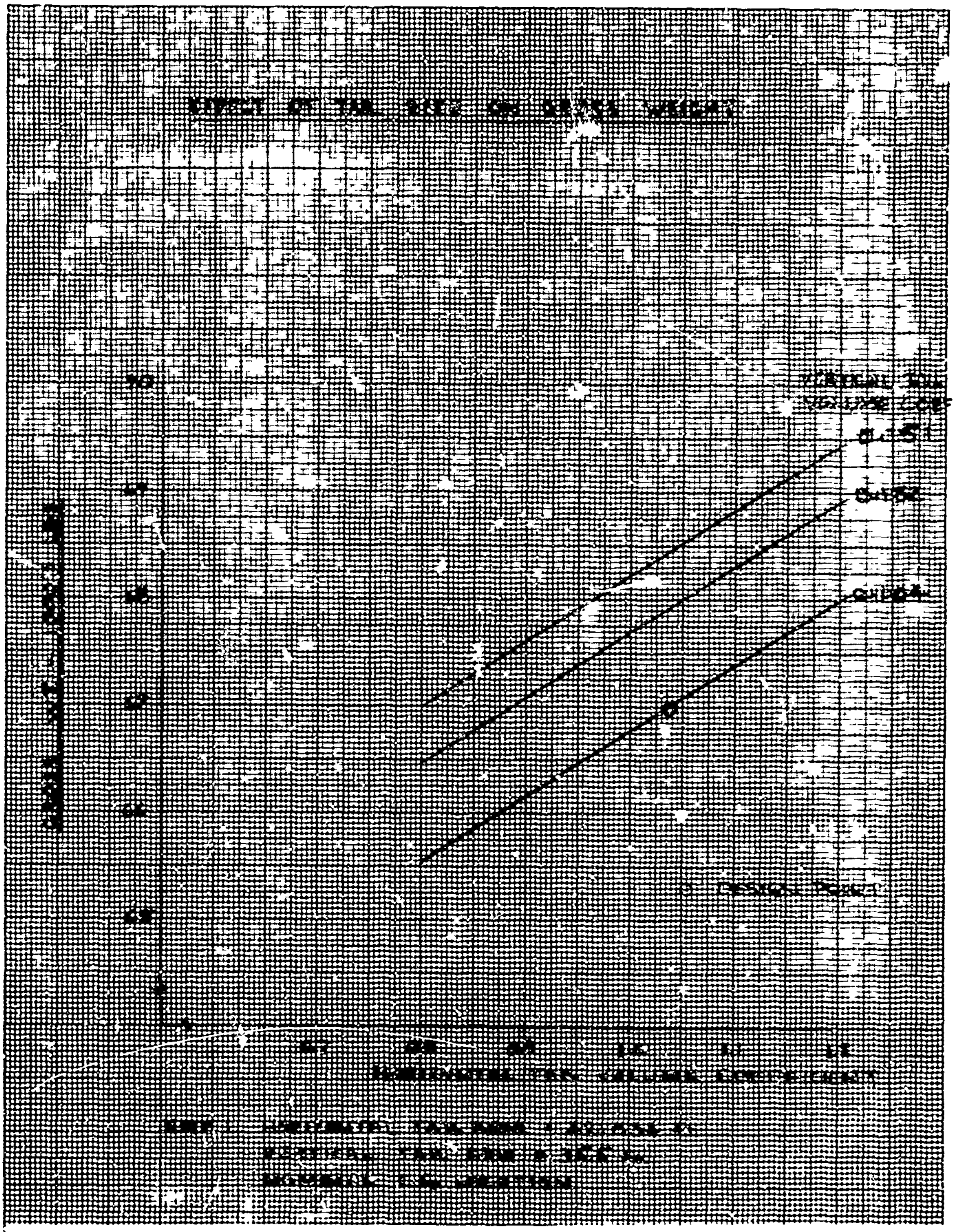

FIGURE III-12 VARIATION OF MODEL 215 GROSS WEIGHT WITH HORIZONTAL AND VERTICAL TAIL VOLUME

Since the cruise (dash speed) condition sizes the engine, the gross weight of the vehicle is extremely sensitive to parasitic drag. This trend curve is given in Figure III-14 for the design point vehicle. In the Phase II effort the reduction of drag, especially at high Mach Nos. will be a major concern and high Mach No. airfoils sections (using Pearcly's "peaky" criterion) presently under investigation at Boeing will be evaluated.

The effect of wing loading is complicated by the unusual wing load bearing requirements. For minimum weight the wing span is dictated by the rotor radius and clearance from the fuselage. Thus changes in wing loading are effective chord variations. The tilt rotor wing is required to carry the full rotor thrust at the wing tips and hub moments and shears transmitted to the wing require the use of a large wing structure box. The effect of wing loading on gross weight as shown in Figure III-15.

In order to provide visibility in selecting engines, a curve of gross weight to a reference specific fuel consumption is given in Figure III-16. The Lycoming LTC4V1 is in the SFC* of 0.42 class which is the design point marked on the curve.

The combination of low disc loading and high dash speed provides excess power for the initial version aircraft in hover. The

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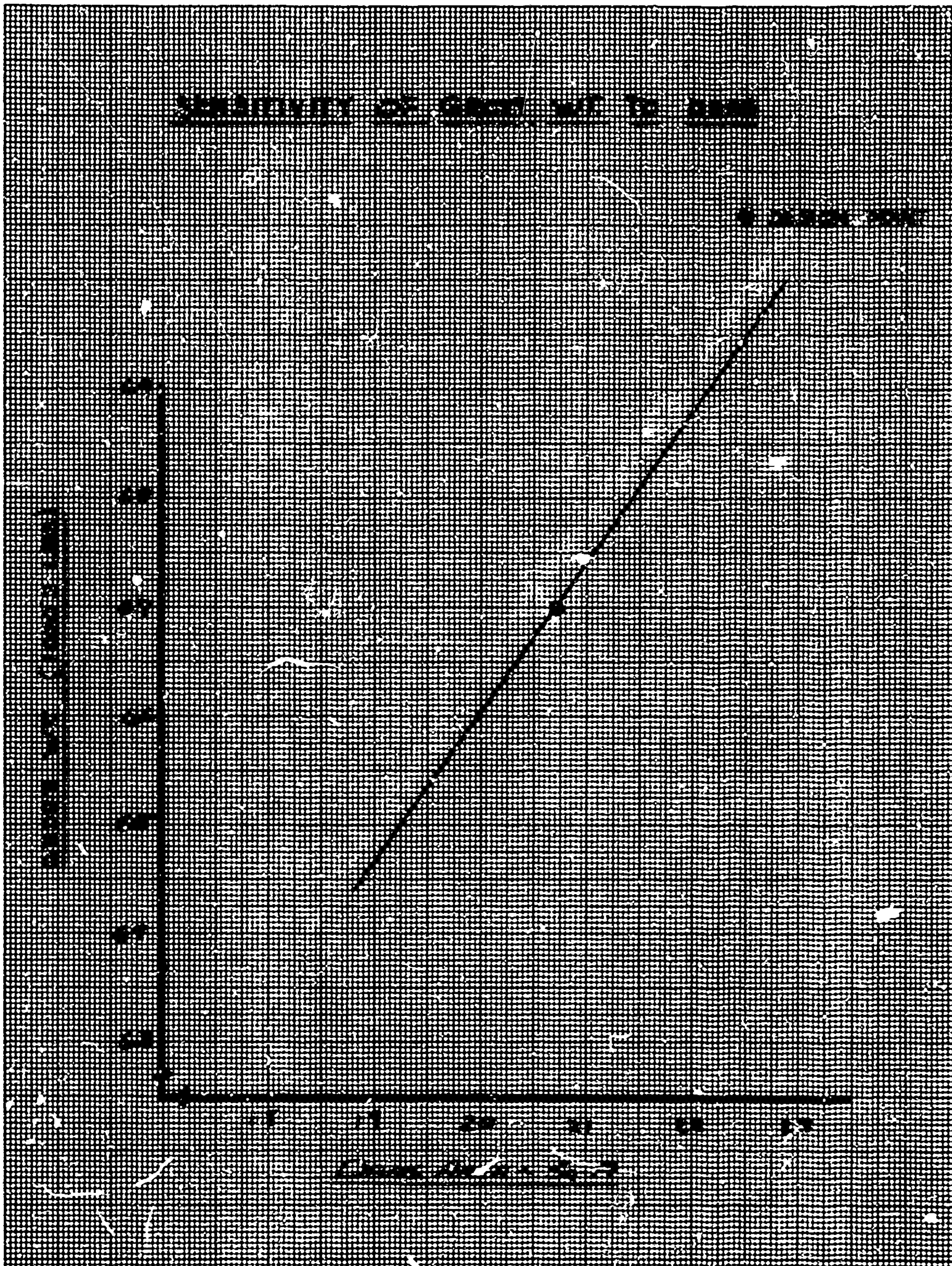


FIGURE III-14 VARIATION OF MODEL 215 GROSS WEIGHT WITH AIRCRAFT DRAG

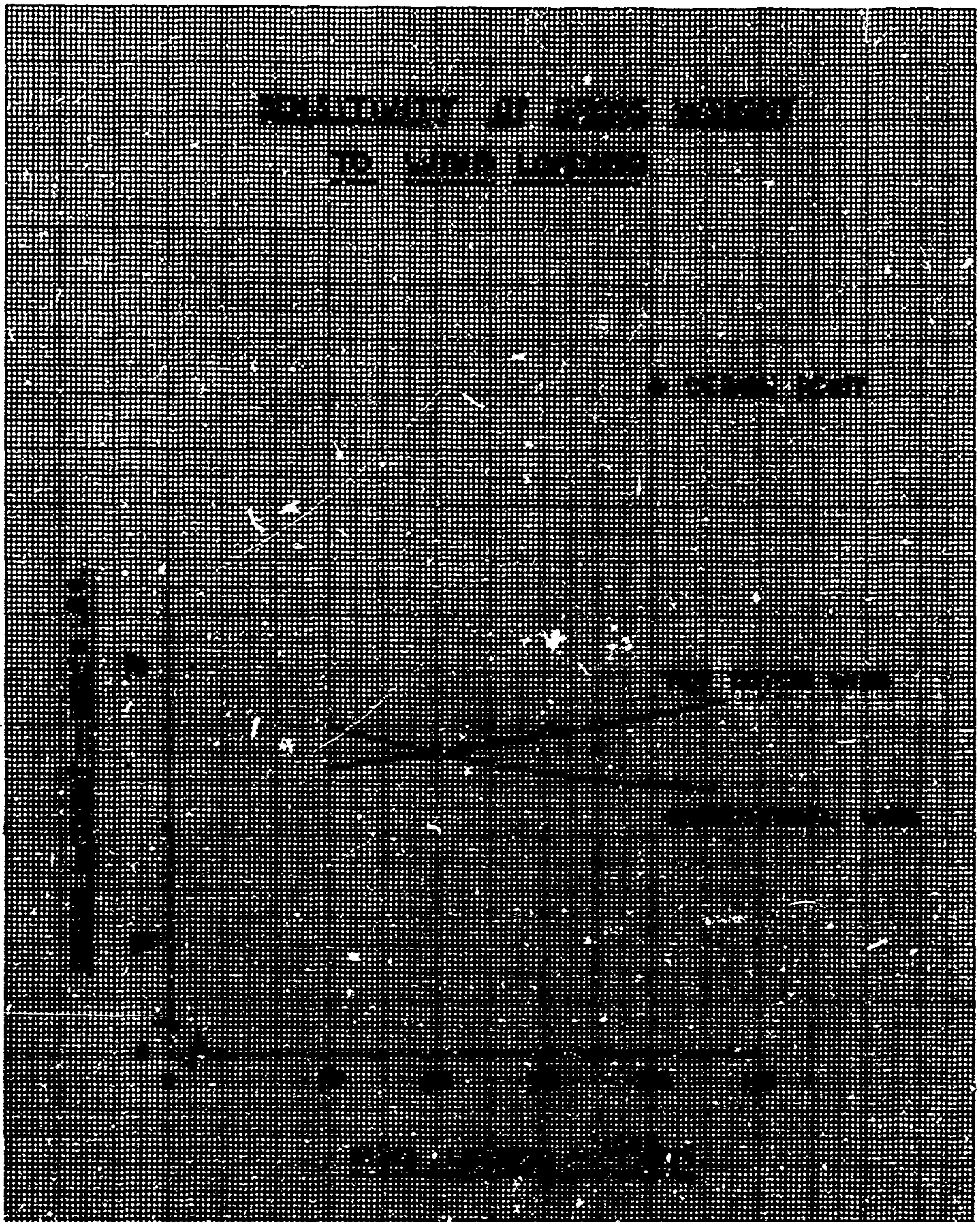


FIGURE III-15 VARIATION OF MODEL 215 GROSS WEIGHT WITH WING LOADING

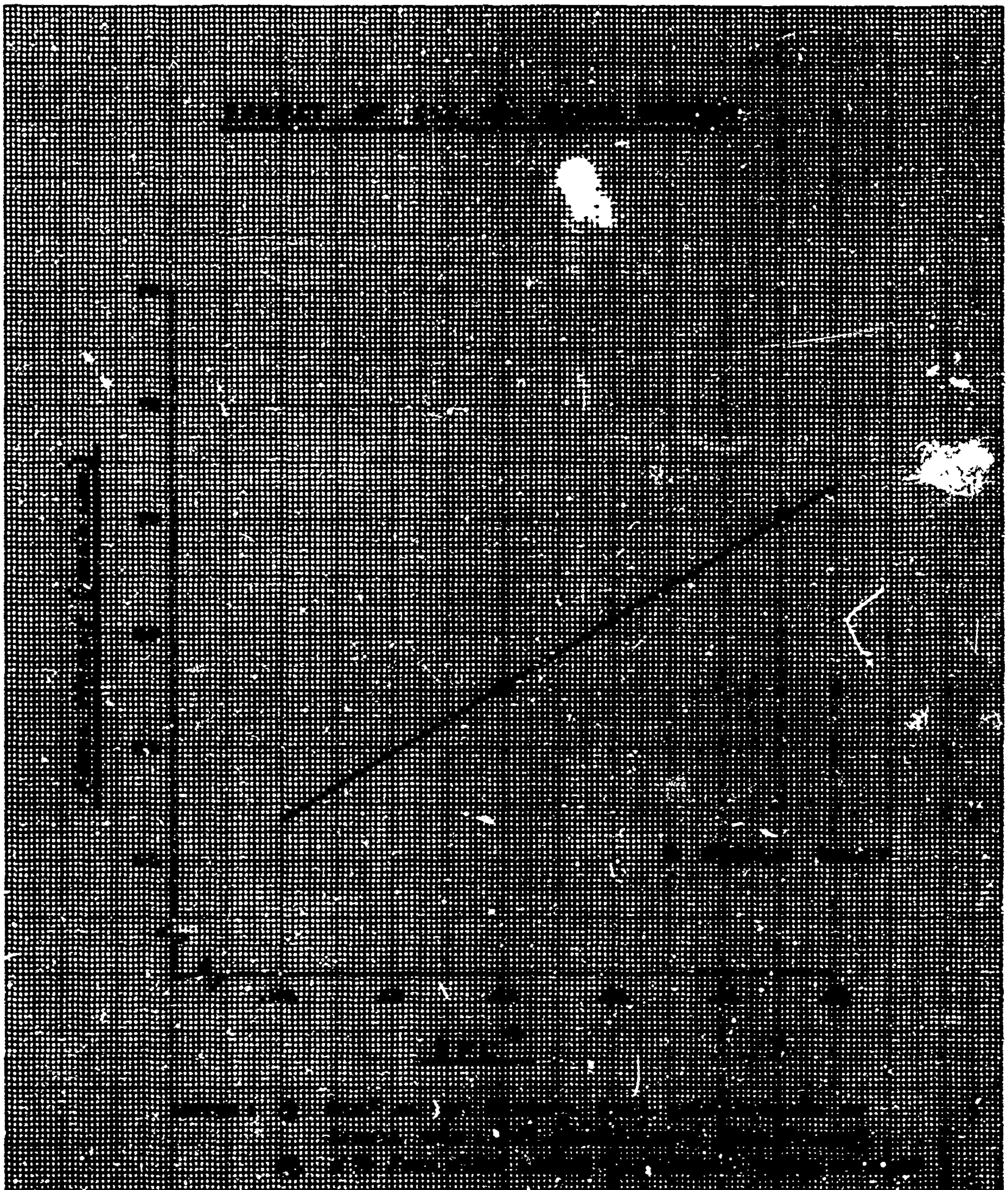


FIGURE III-16 VARIATION OF MODEL 215 GROSS WEIGHT WITH ENGINE SPECIFIC FUEL CONSUMPTION data presented in Figure III-17 gives the sensitivity of gross weight to the dash speed requirement. As the aircraft matures in production, the gross weight (and payload) will increase. This requires a larger increase in power to maintain the same hover performance than is required to maintain the same dash speed. Therefore, as the aircraft grows it will approach a power match at vertical and horizontal flight conditions. This curve demonstrates that in order to provide a power match at the prop/rotor diameter selected the dash speed would have to be reduced to 270 kt (TAS) at 10000 ft standard.

The download allowance (T/W = 1.043) and hover maneuver margin (1.15g) used in the design of Model 215 are discussed in detail in Section IV. In order to provide insight into the weight penalty incurred by increased maneuver load factor capability or download weight sensitivity, data is shown in Figure III-18. The effect of cruise efficiency on the design gross weight is shown in Figure III-19. This reflects the importance of accurate cruise efficiency prediction.

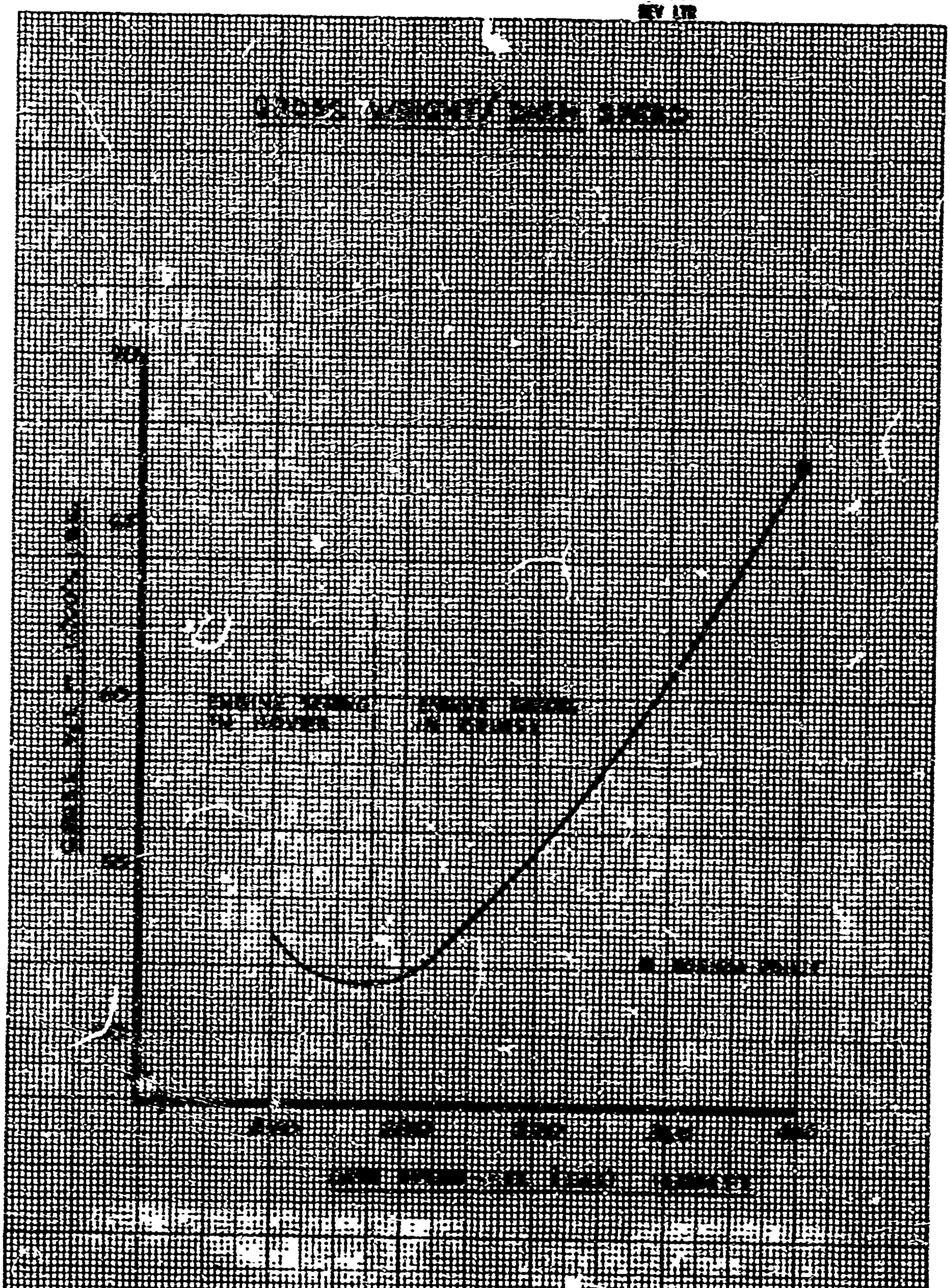
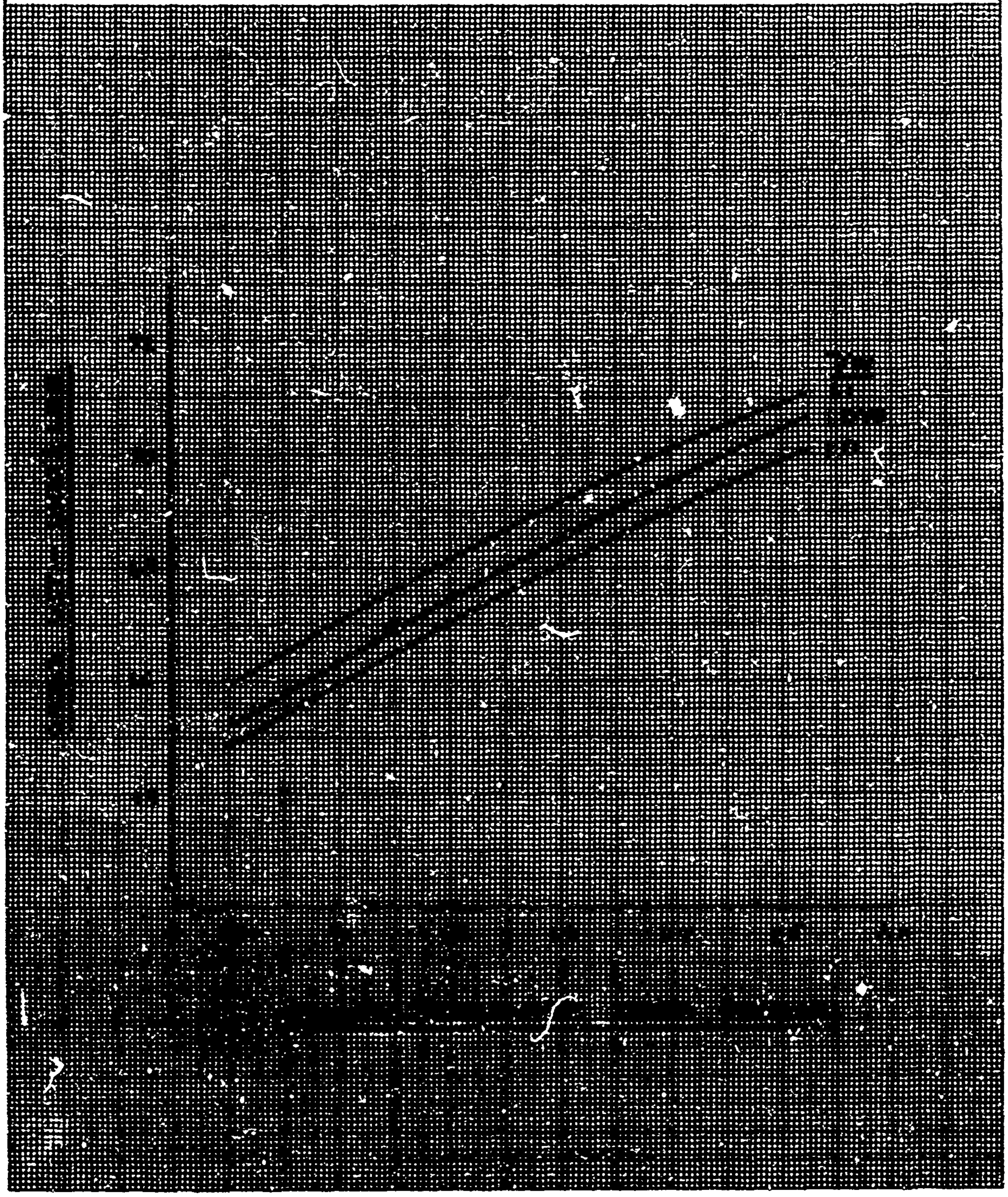
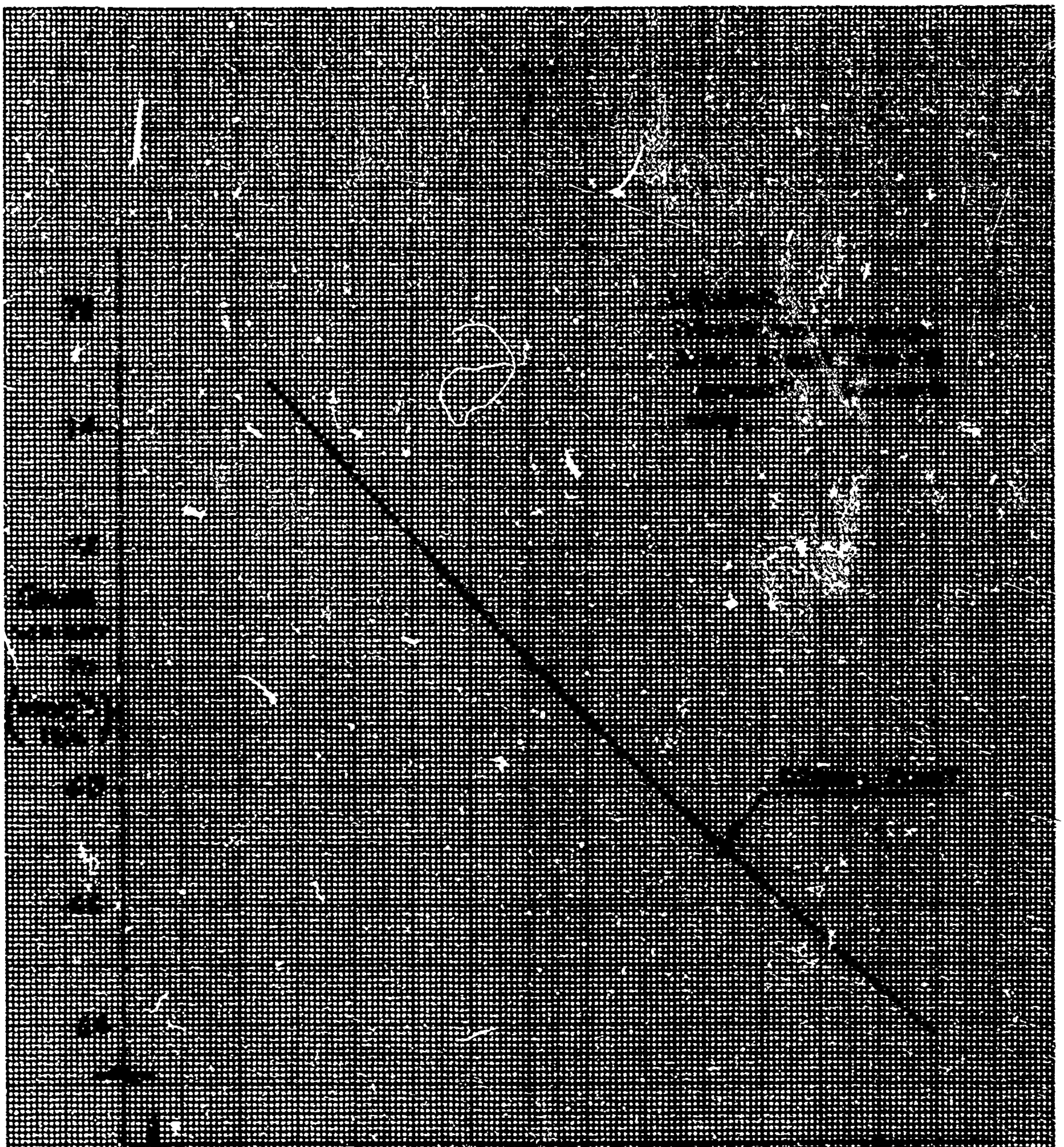


FIGURE III-17 VARIATION OF MODEL 215 GROSS WEIGHT WITH REQUIRED DASH SPEED

FIGURE III-18 SENSITIVITY OF GROSS WEIGHT TO DOWNLOAD AND MANEUVER LOAD FACTOR





SECTION IV

MISSION PERFORMANCE

1. SUMMARY

The performance task of the tilt/rotor configuration is one of design compromise between the two flight modes of hover and cruise flight. The design end points of hover time, altitude, temperature and weight coupled with the high speed cruise conditions define a particular vehicle.

For the baseline configuration with a dash speed of 400 knots (TAS) at 10,000 ft.,standard,and hover requirements at 2,500 ft.,93° the emphasis of the design is weighted heavily on the cruise condition. This does not mean that hover problems can be neglected since small increments of hover download or rotor efficiency cause sizeable increases in the design gross weight.

In this section of the report, the performance for the transport mission, the rescue mission, and the alternate transport mission are discussed. Then the methodology used for hover and cruise is discussed. Transitional performance and STOL take-off are discussed as applicable to the mission performance.

2. TRANSPORT MISSION

Aircraft sizing and performance for the transport mission shown in Figure IV-1 has been estimated by using the V/STOL Aircraft Sizing and Performance Computer Program (VASCOMP II), Reference IV-1. The baseline Model 215 configuration (GW = 67,000 lbs.) described in Section III was sized to fly the primary transport mission. The mission performance fuel requirements are given in Table IV-1 and the mission time history is plotted in Figure IV-2. The design gross weight of this aircraft is 67,000 lbs and 10,224 lbs. of fuel are required to fly the basic transport mission with a payload of five tons. The total mission time is 1.7 hours.

A. Hover

The hover performance for the transport mission is shown in Figure IV-3. These calculation are based on a download of 4.3% of the hover gross weight, an altitude of 2,500 feet at 93° and rotors sized to provide a net thrust load factor of 1.15 in hover before the stall flutter rotor limit is reached. Rotor limits, load factor and download are discussed in the following sections on hover methodology. The hover RPM for this condition is 295.

TILT ROTOR TRANSPORT MISSION PERFORMANCE

FIGURE IV-1 SUMMARY OF TRANSPORT MISSION PERFORMANCE REQUIREMENTS

[illegible]

TABLE IV-1

BASELINE CONFIGURATION TRANSPORT MISSION

T.O. Gross WT = 67,000 lbs. (5% Fuel Allowance)

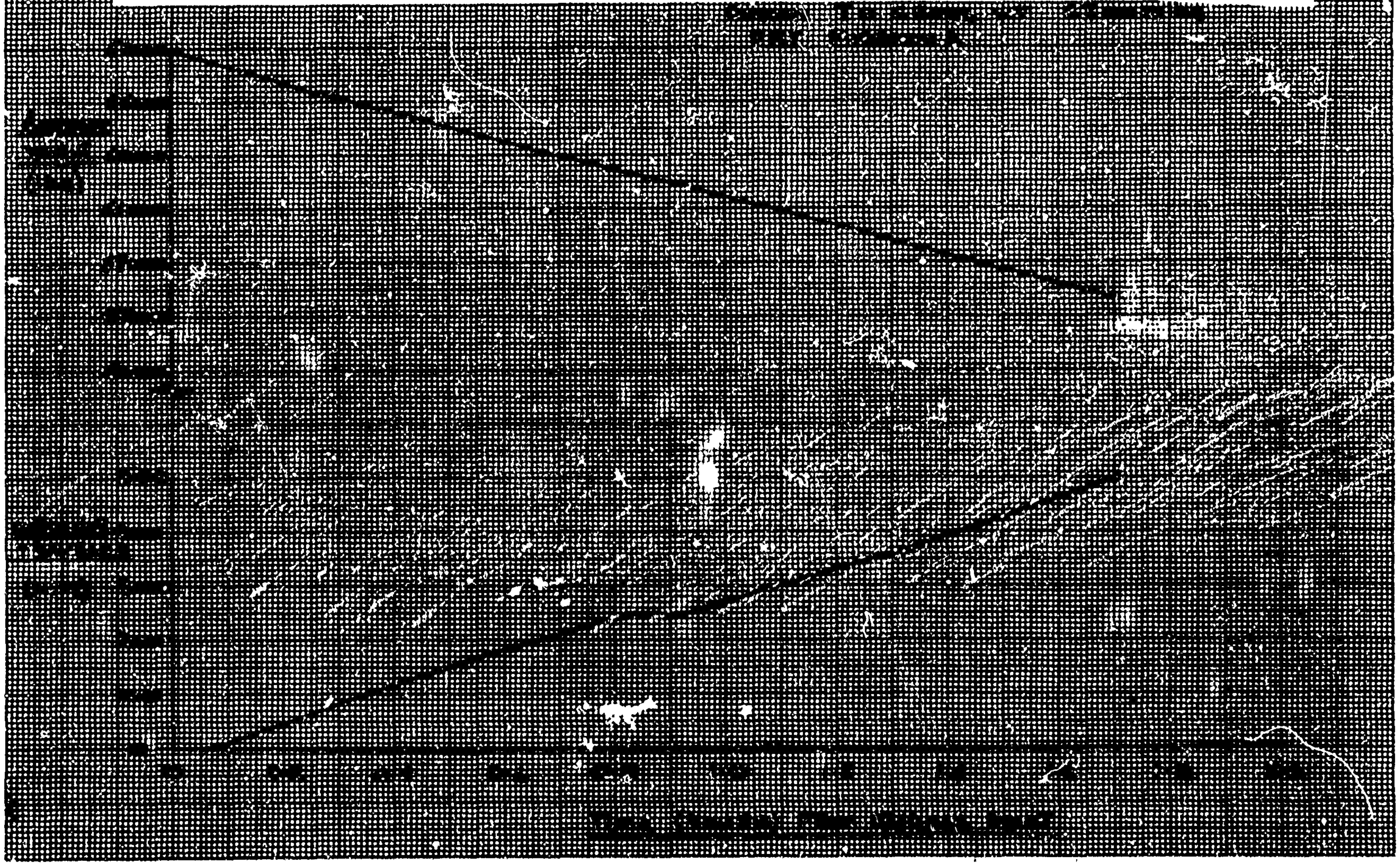
Segment	Altitude (Ft)	Temp °F	Range N.M.	Mean Airspeed TAS (kts)	Fuel (lbs) Used (End of Segment)	Mean Spec Range (N.M./#)
Warm Up-Taxi	0.0	Std. Day	0.0	0.0	320	N.A.
T.O. and Hover	2,500	93°	0.0	0.0	313	N.A.
Climb	2,500 to 10,000	Std. Day	6.85	210	559	N.A.
Cruise	10,000	Std. Day	150.0	350	2,856	0.0623
Cruise	0.0	Std. Day	250.0	300	4,622	0.0566
Hover Land	2,500	93°	250.0	0.0	4,794	N.A.
.change Payload	2,500	93°	250.0	0.0	4,794	N.A.
Warm Up-Taxi	0.0	Std. Day	250.0	0.0	5,015	N.A.
T.O. and Hover	2,500	93°	250.0	0.0	5.101	N.A.
Cruise	0-0	Std. Day	350.0	300	6,848	0.0573
Climb	0-0 to 10,000	Std. Day	356.7	206	7,102	N.A.
Cruise	10,000	Std. Day	500.0	350	9,295	0.0654

Mission Fuel Required 9,295 Lbs.

10% Reserve Fuel 929 Lbs.

TILT ROTOR TRANSPORT MISSION PERFORMANCE

FIGURE IV-2 TIME HISTORY OF FUEL BURN OFF AND DISTANCE TRAVELED DURING TRANSPORT MISSION



The selection of the number of engines is influenced markedly by the hover performance with an engine out. The requirements for engine out conditions are that the aircraft will have sufficient power to convert to the cruise mode or return safely to the ground. The power available at maximum power setting on standard and 93^{o} days for three of four engines operating and for one of two engines operating is compared with power required in Figures IV-3 and 4. At the design mission requirement of 2,500' 93^{o}, it is shown that the two engines, one of which is inoperative, the hover requirements cannot be met at take-off or mid-mission gross weight. With a four engines configuration, all hover conditions for both the transport and the rescue missions can be met with three engines operating at less than max. power. This consideration is a major factor in the decision to provide a four engine (two per pod) aircraft.

The hover performance of the rotors for the transport aircraft is given in Figures IV-5 and IV-6 and indicates a peak figure of merit of 71.8% at the design thrust coefficient of 0.0718 (0.009175 in rotor notation). It should be emphasized that this hover performance level is compromised to provide

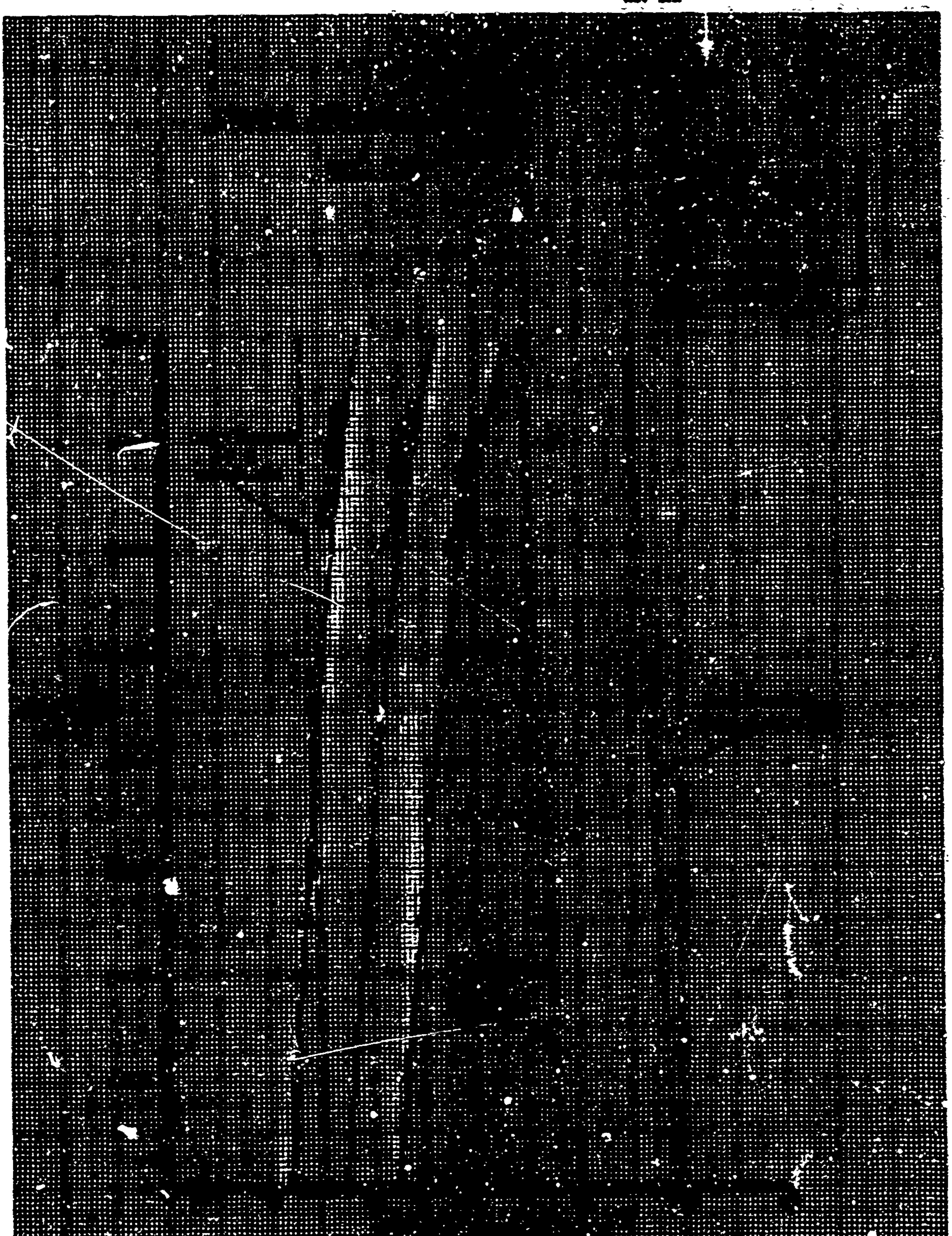


FIGURE IV-3 MISSION (HOT DAY) HOVER PERFORMANCE SHOWS ADEQUATE POWER MARGIN FOR NEAR HOVER MANEUVERS WITH ONE OF FOUR ENGINES IN- OPERATIVE

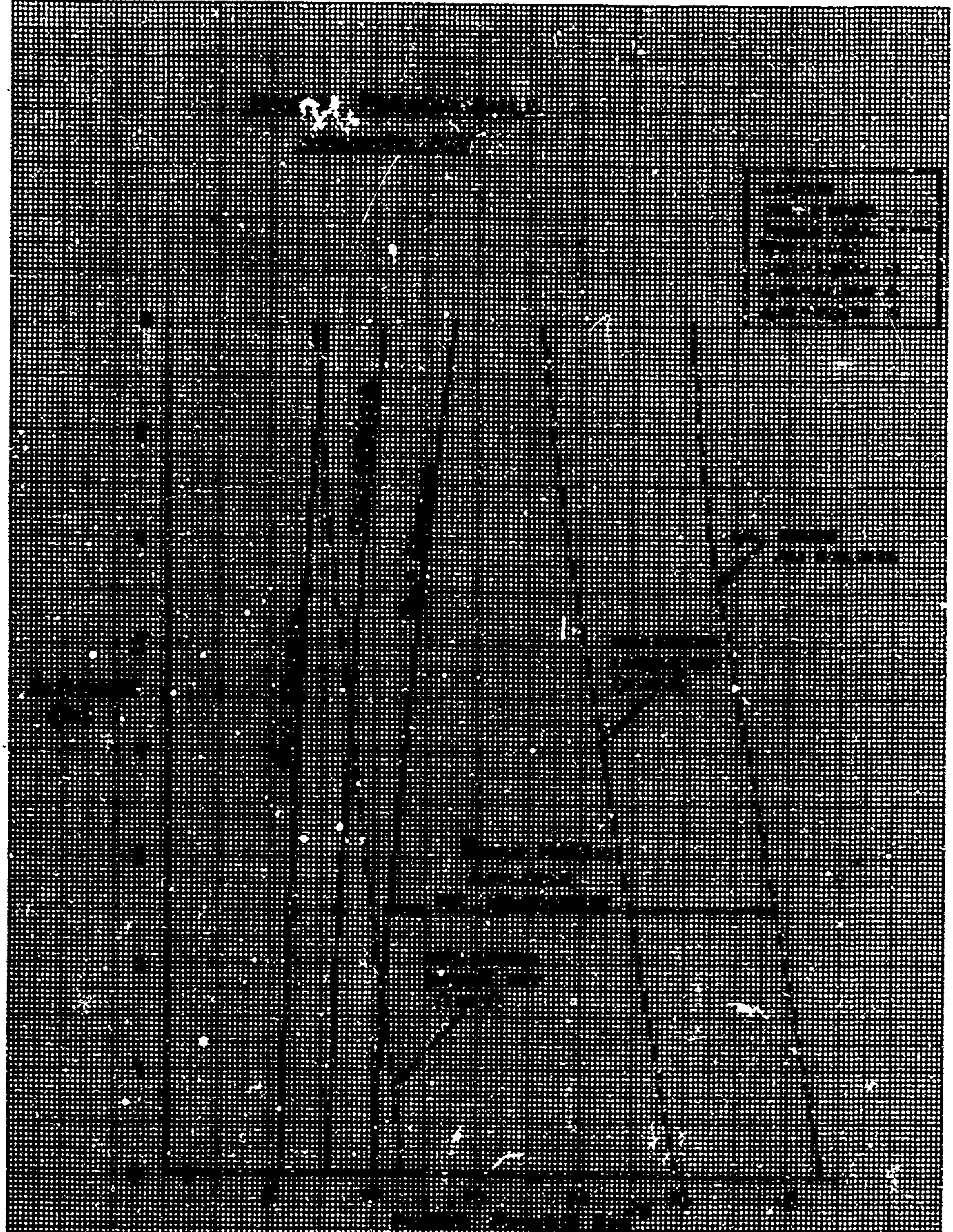


FIGURE IV-4 HOVER PERFORMANCE (STANDARD DAY) SHOWS LARGE POWER MARGIN FOR NEAR HOVER MANEUVERS AND ADVANTAGE OF 4 ENGINES FOR ENGINE-OUT CASE

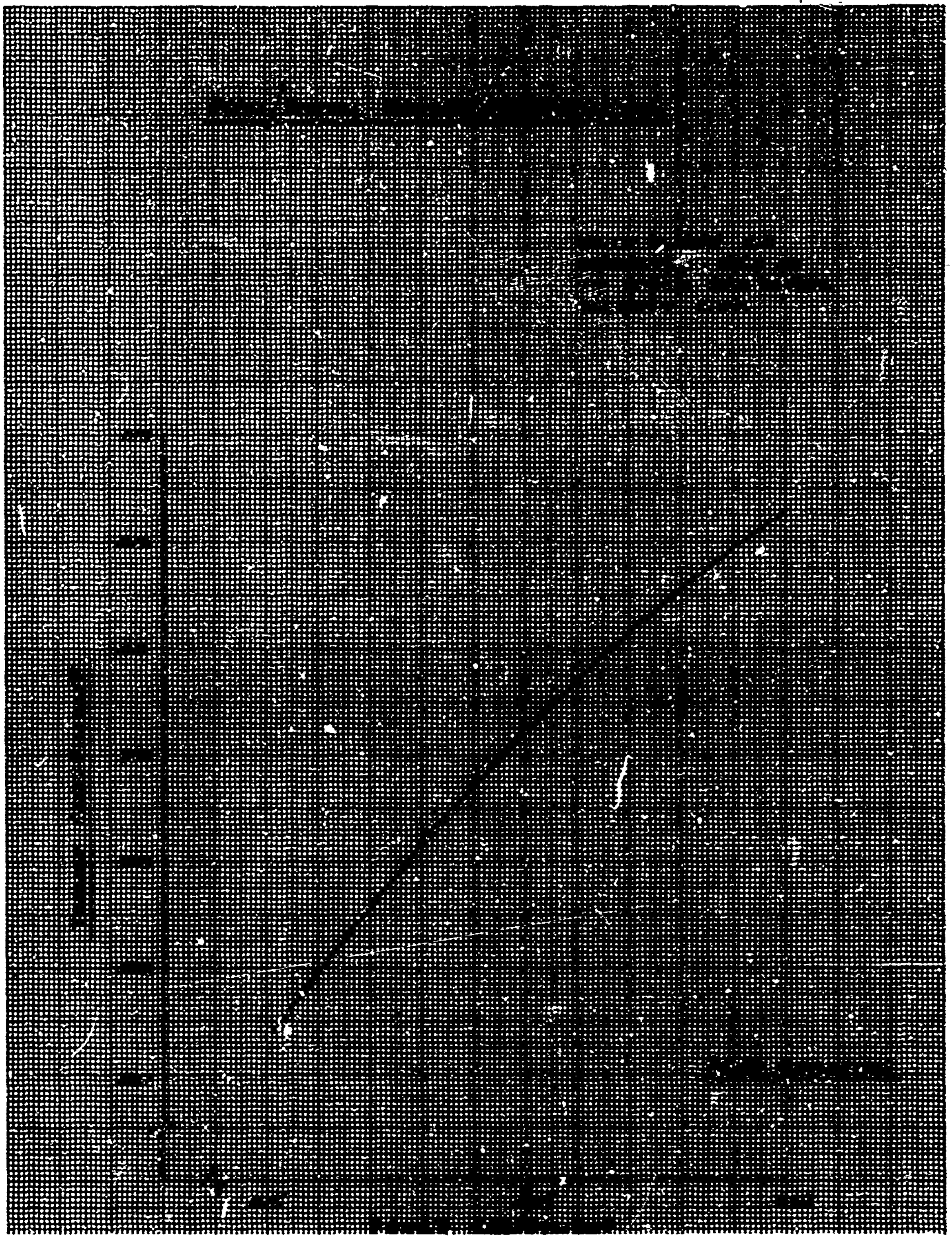


FIGURE IV-5 PREDICTED HOVER THRUST AND POWER REQUIRED CO-EFFICIENTS FOR ROTORS OF MODEL 215 AIRCRAFT

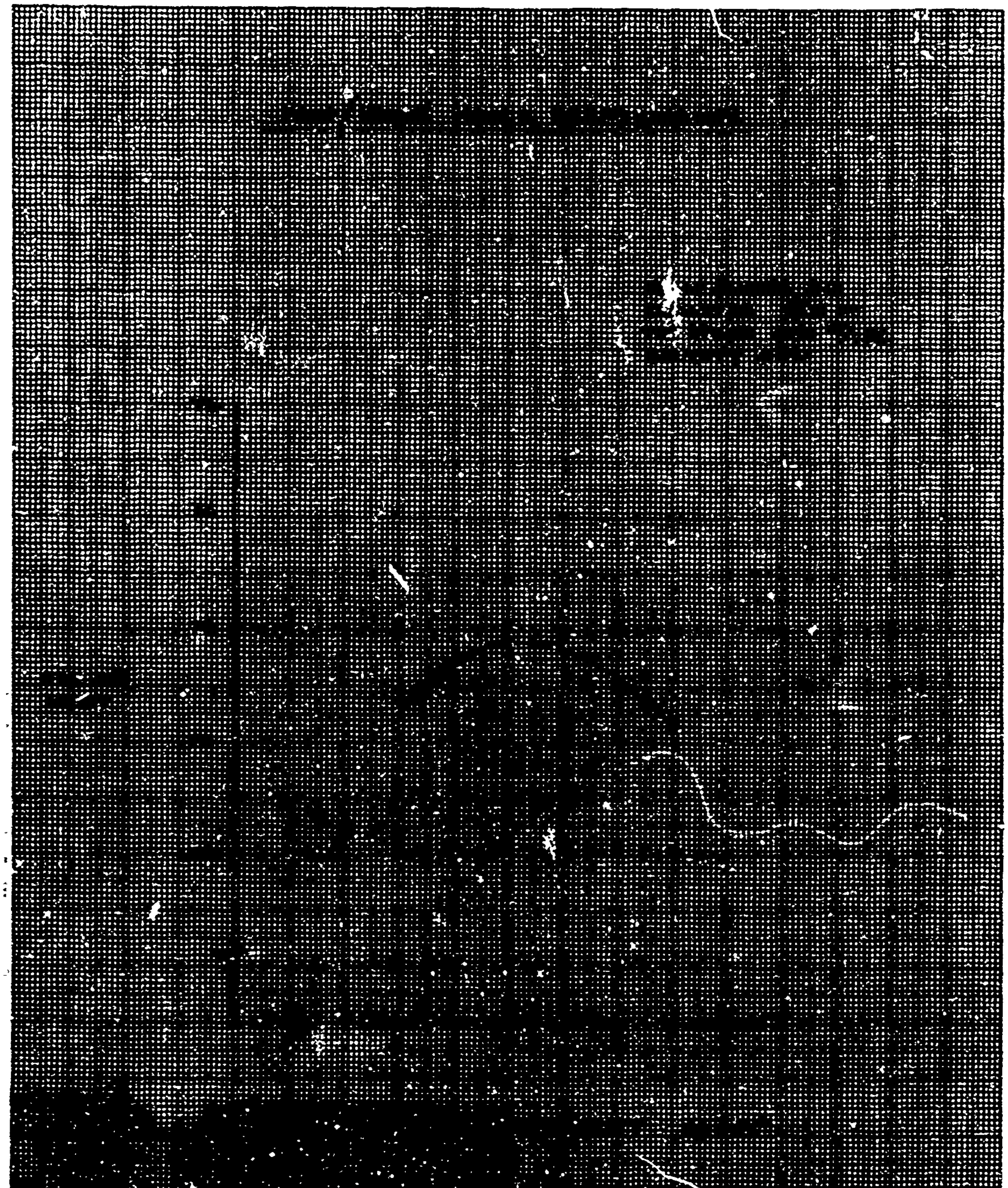
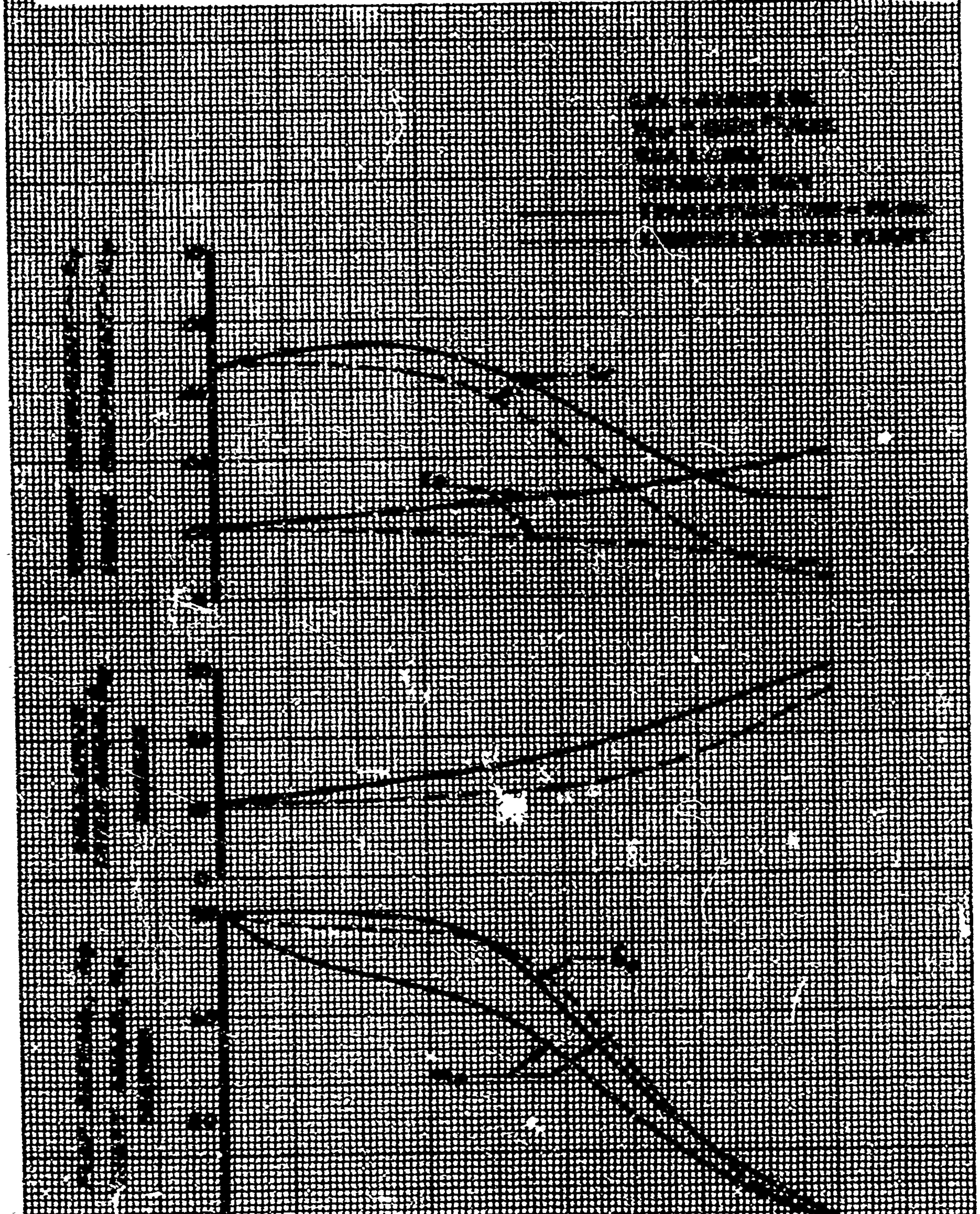


FIGURE IV-6 HOVER FIGURE OF MERIT OF THE ROTORS DESIGNED FOR THE MODEL 215 AIRCRAFT an optimum trade-off with cruise efficiency and thus minimize the gross weight of the aircraft. A major task in the Phase II of this work will be to expand this optimization to systematically include stress, weight and dynamics limitations. Further developments in high Mach number blade sections currently under investigation at Boeing can also be incorporated at that time.

B. Transition

The preliminary design of the transport configuration is primarily considered at the end points of the flight envelope (hover and cruise). A constraint on the design is the maintenance of an acceptable transition corridor. Such performance is estimated in level flight and accelerated transition characteristics as shown in Figure IV-7. The accelerated transition shown is completed in 24 seconds from hover to 180 knots with an average acceleration of 0.4g. In the early stages of transition, the umbrella flaps are open to minimize download due to prop/rotor downwash. The umbrella flaps are kept open up to a velocity of approximately 50 fps in order to provide a wing spoiler action. This is to ensure that both wing lower surfaces unstall at the same time (when the umbrella is closed).

FIGURE VI-7 TYPICAL TRANSITION SCHEDULES FOR MODEL 215 AIRCRAFT



The transitional data presented in this section is based on wind tunnel test data of an unpowered model, Reference VI-3 and preliminary data from the 13 ft. Dia. Model 215 isolated rotor tests conducted at ONERA this year. Typical transition rotor performance characteristics are given at a propeller advance ratio (J) of 0.4 in Figure IV-8.

C. <u>Climb</u>

The transport mission requires normal rated power climbs with all engines operating at cruise rpm in standard day conditions. As shown in Figure IV-9, this aircraft will climb at rates greater than 3,500 feet per minute under these conditions. The maximum rate of climb at sea level for a 67,000 lb aircraft is 4,561 ft/minute and the indicated service ceiling is 26,000 ft. This performance is calculated using standard airplane methodology. Performance of the aircraft under engine-out conditions is shown in Figures IV-10 and IV-11 .

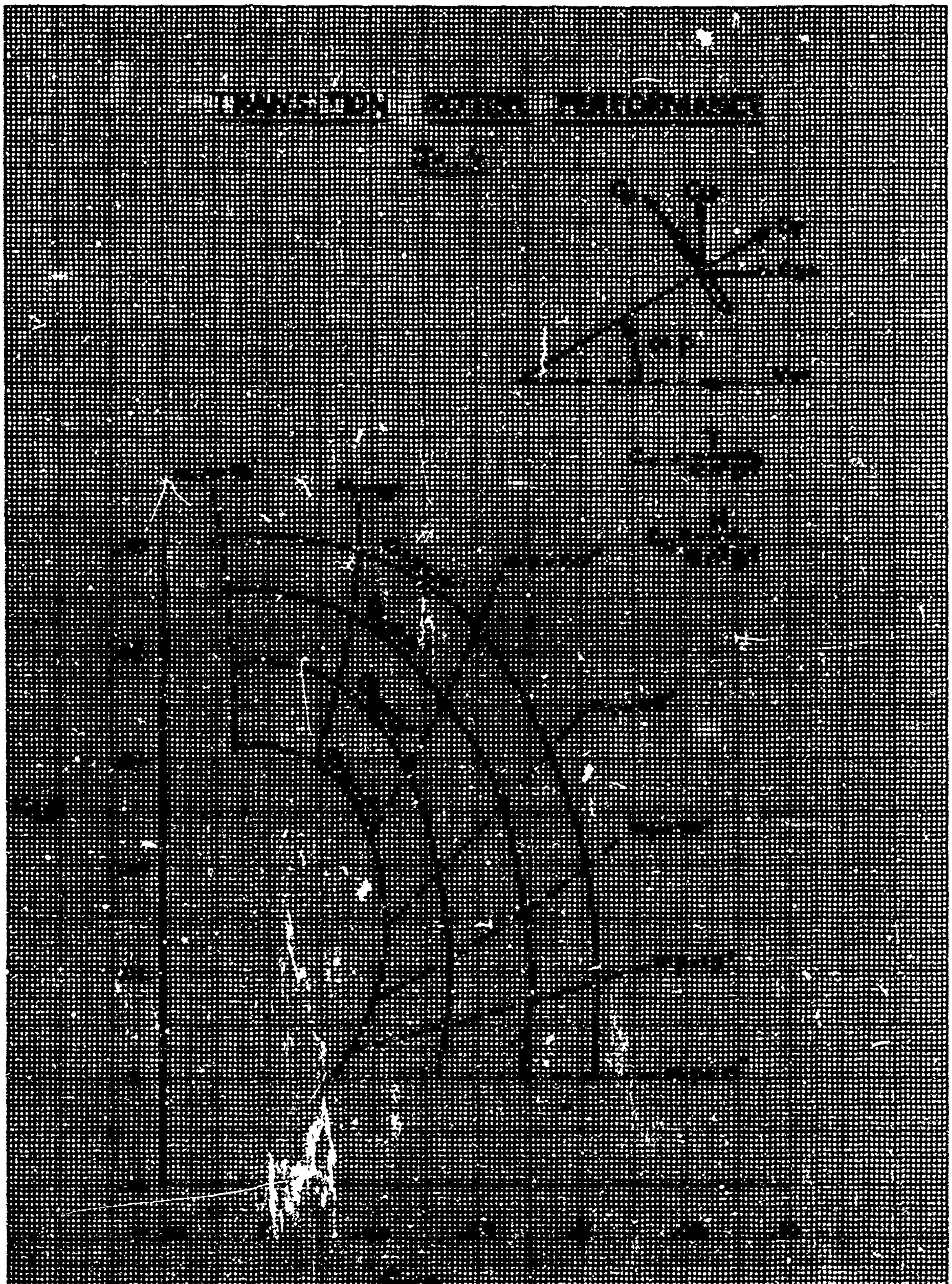


FIGURE IV-8 TYPICAL PROPELLER PERFORMANCE DATA USED IN TRANSITION ANALYSIS

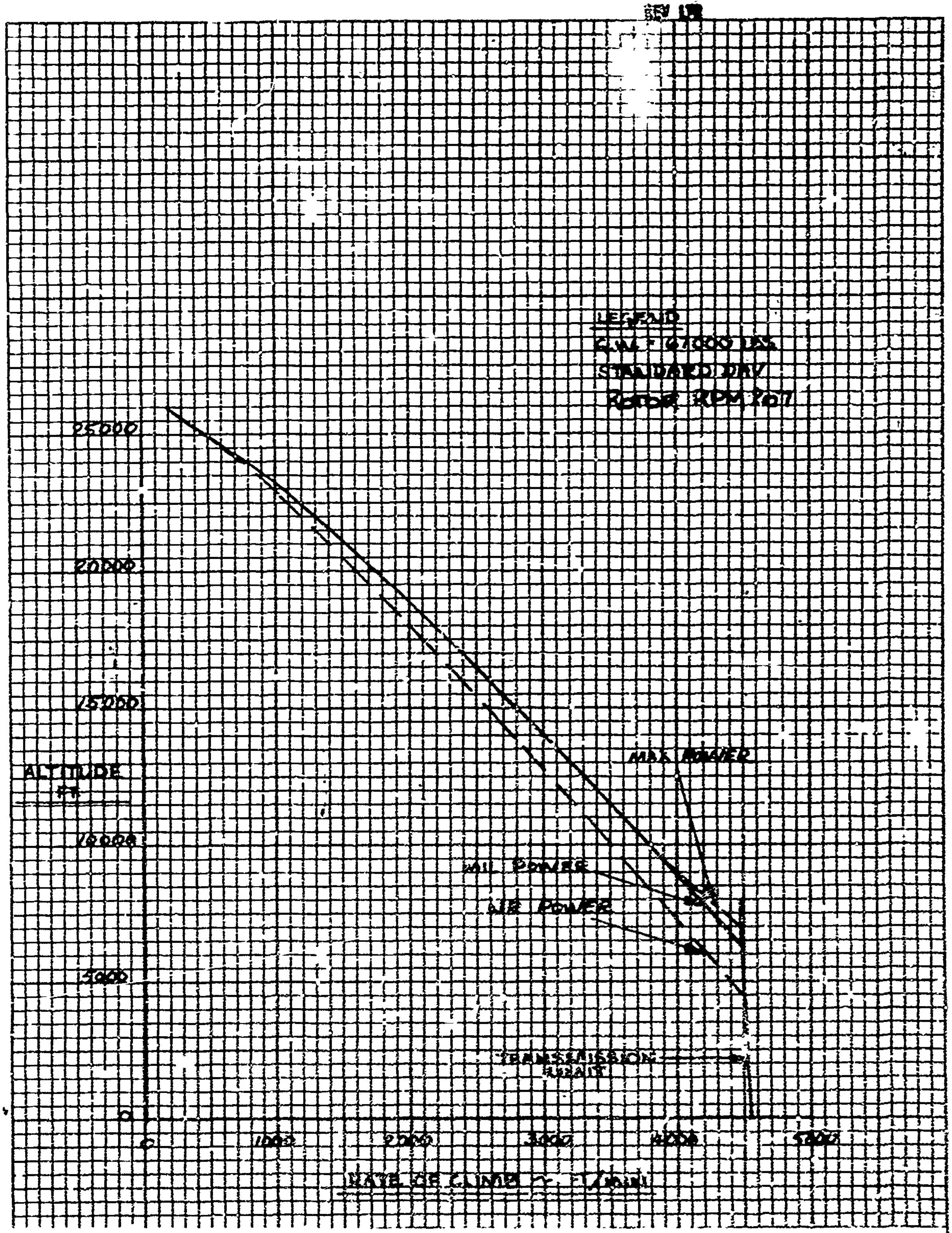


FIGURE IV-9 MAXIMUM RATE OF CLIMB PERFORMANCE OF TRANSPORT AIRCRAFT IN AIRPLANE MODE

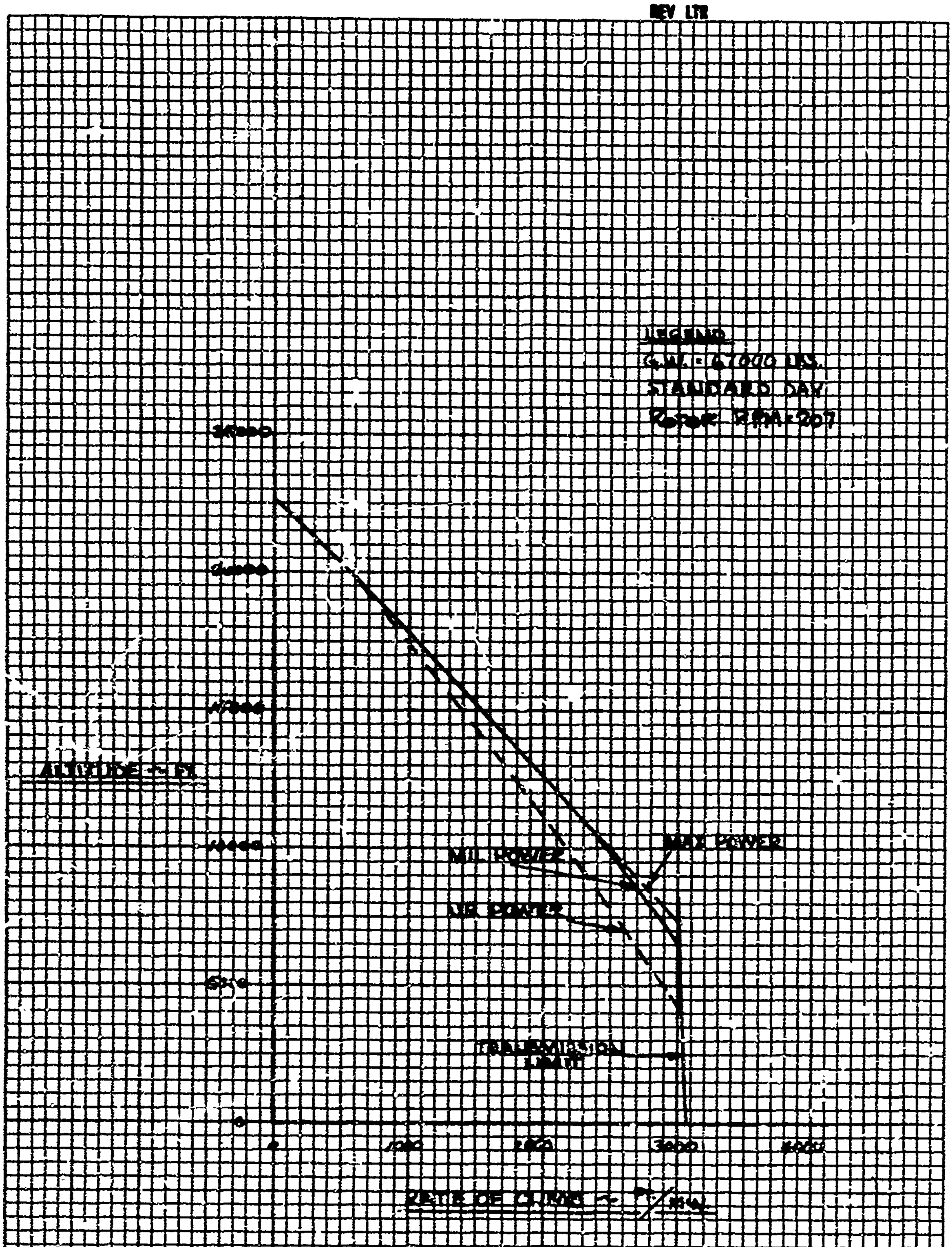


FIGURE IV-10 MAXIMUM RATE OF CLIMB OF TRANSPORT AIRCRAFT WITH ONE OF FOUR ENGINES INOPERATIVE

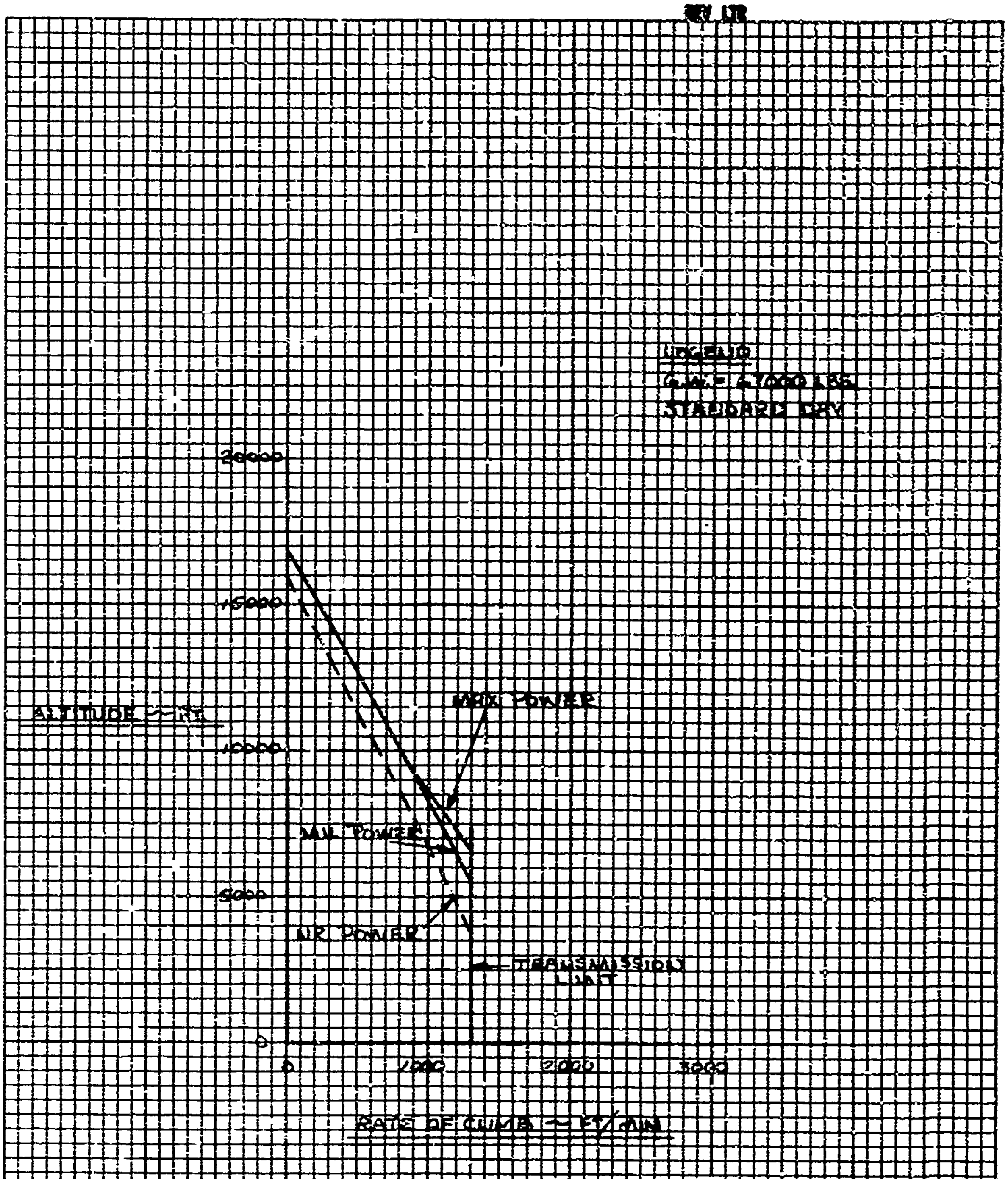


FIGURE IV-11 MAXIMUM RATE OF CLIMB OF TRANSPORT AIRCRAFT WITH ONE ENGINE OUT FOR TWO ENGINE CONFIGURATION

D. <u>Cruise and Dash</u>

The cruise and dash performance of the flight vehicle are critical for two basic reasons. First, the dash speed requirement of 400 knots sizes the engines and installed horsepower. Secondly, the cruise performance dictates the pay-load-range qualities of the aircraft and as such defines it productivity. These considerations require the design emphasis to be placed in the airplane mode to derive the lightest gross weight.

The particular problem areas are the minimization of airplane parasite drag and the maximization of prop/rotor efficiency. These requirements are constrained by the weight and stress constraints of wing design where a thick sectioned low aspect ratio wing (21% thickness ratio and AR = 5.2) has been selected based on the requirement to maintain a sufficiently low stall speed to provide an adequate transition corridor with a simple flap system.

The baseline transport configuration (GW = 67,000 lb) power required and available curves are shown in Figures IV-12 and IV-13. These calculations are based on the airplane drag data given in Section VI and engine performance calculated as described later in this section.

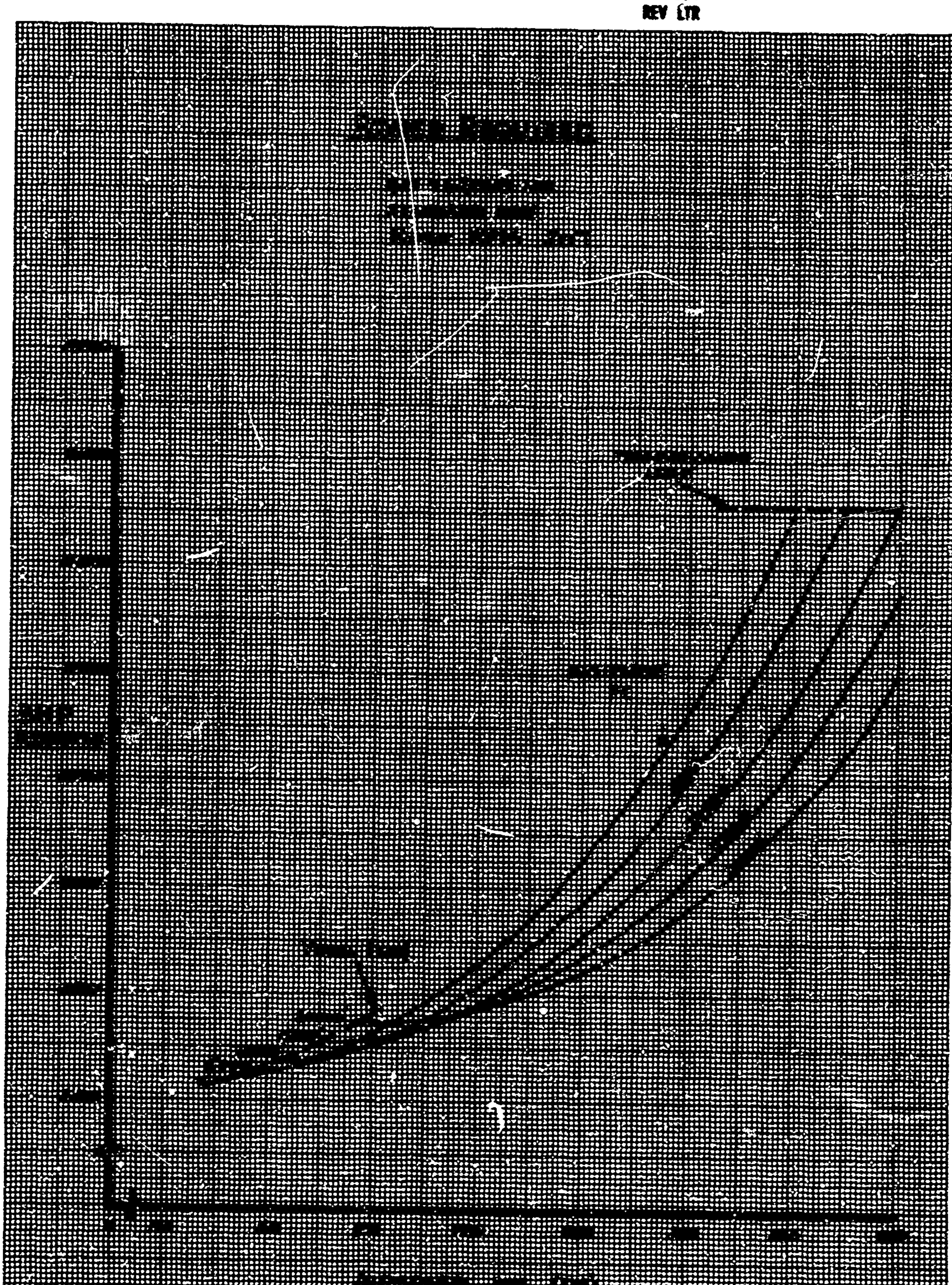


FIGURE IV-12 POWER REQUIRED FOR TRANSPORT AIRCRAFT IN AIRPLANE FLIGHT MODE

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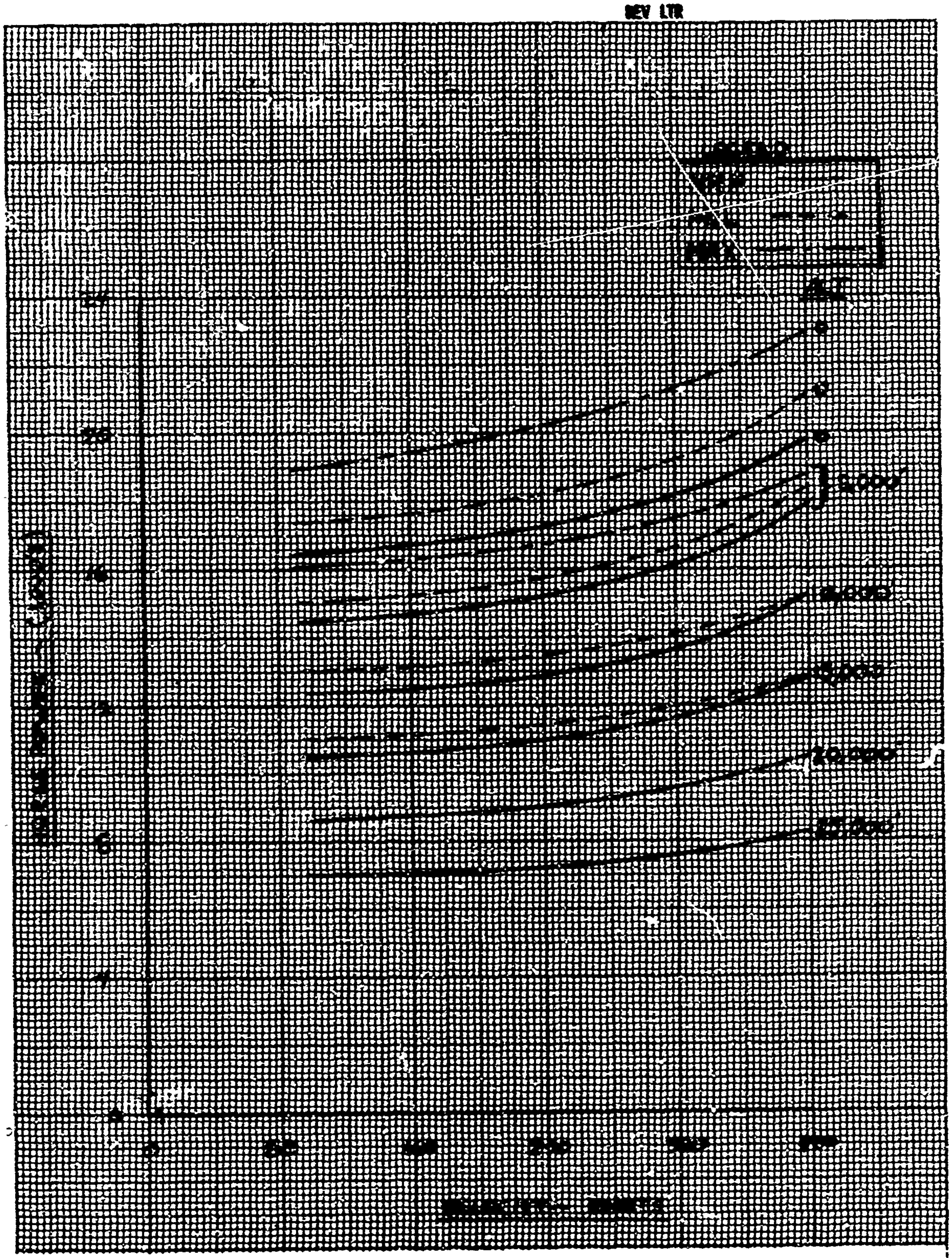

FIGURE IV-13 ENGINE SHAFT POWER AVAILABLE WITH ASSUMED ENGINE CYCLE

It can be noted in Figure IV-13 that at the higher altitudes the power available lines for all power settings (allowable turbine temperatures) coincide. This is due to a primary gas generator rpm limit and is a function of the particular engine cycle chosen for this study.

The prop/rotor cruise efficiency performance used in this study is given in Figure IV-14 and the increase is efficiency with reduced rotational speed is shown in Figure IV-15. The cruise flight prop/rotor RPM is 207 reduced to 70% of the hover value. The sensitivity studies discussed indicate that this reduction ratio is optimum from a minimum gross weight stand point since the increase in cruise efficiency with decreased RPM significantly reduces advance ratio and Mach number effects. The methodology used to calculate propeller efficiency is discussed later in this section of this report.

The intersection of the available and required power lines provide the locus of the maximum steady level flight envelope. These data are given in Figures IV-16 through IV-18 for full power and engine out cases. The impact on cruise velocity of the flat rated transmission is apparent up to 10,000 ft. The maximum velocity lines for all power settings above 10,000 ft. are coincident

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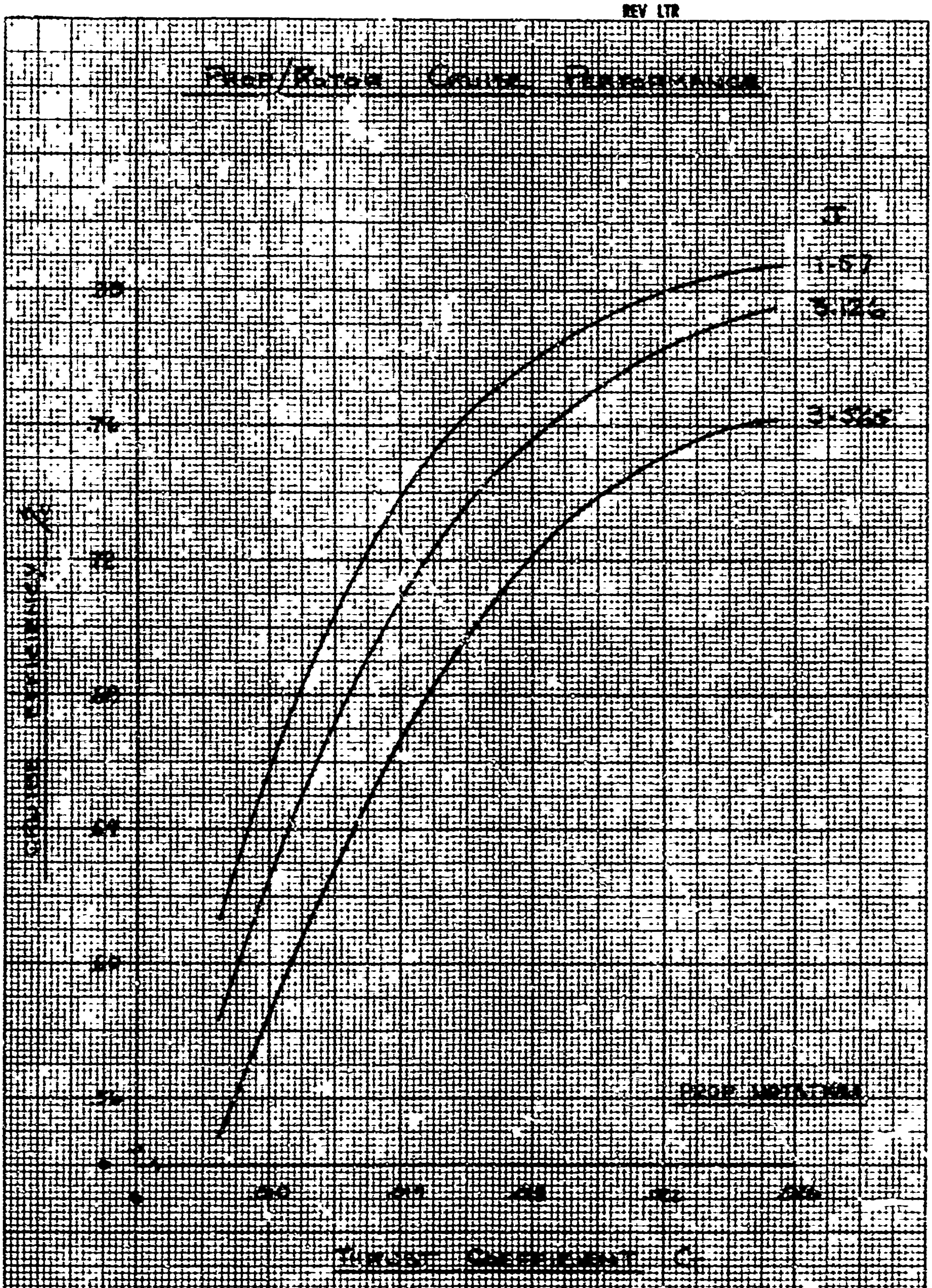

FIGURE IV-14 PROP/ROTOR CRUISE EFFICIENCY PREDICTED FOR ROTOR OPTIMIZED FOR TRANSPORT MISSION

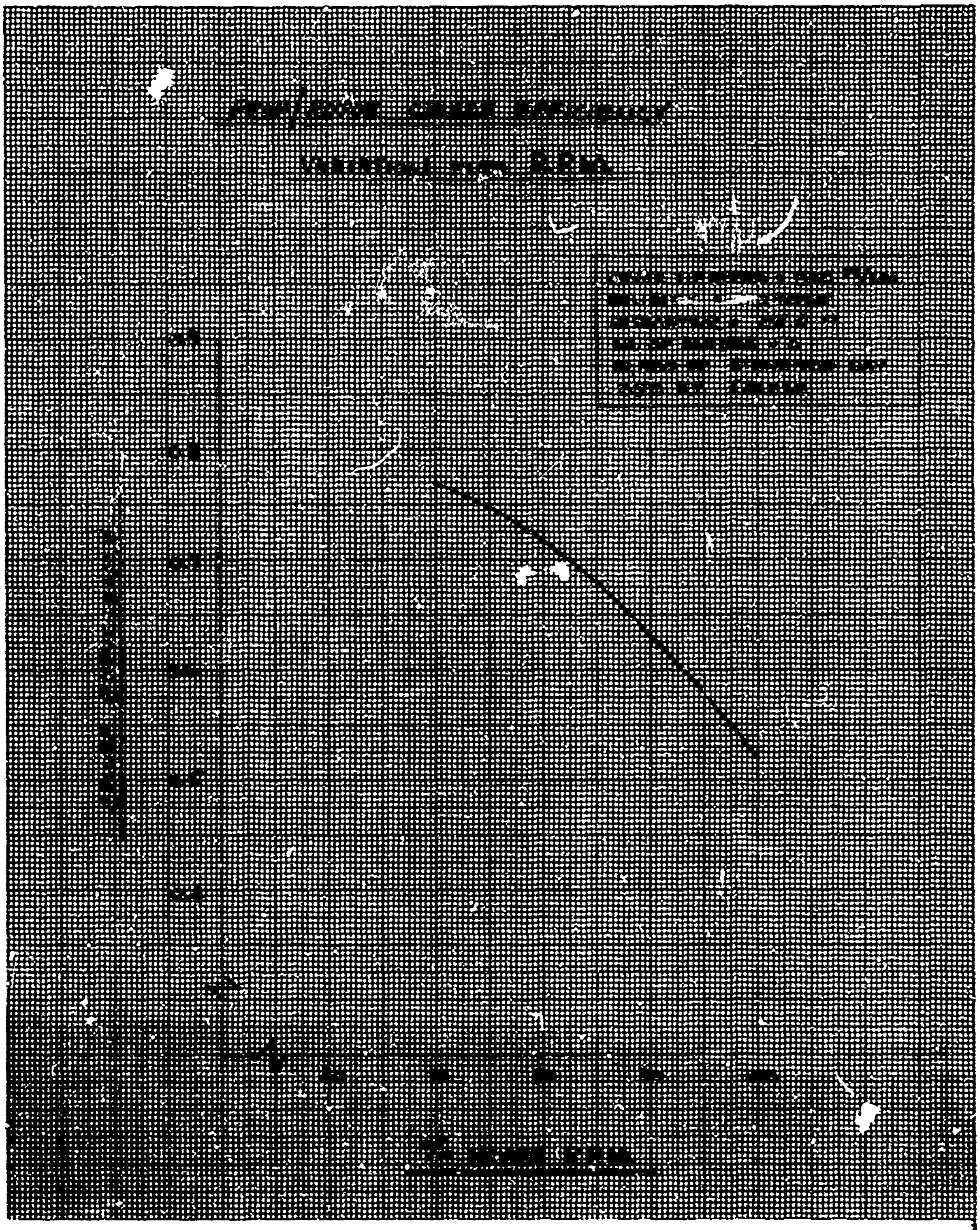

FIGURE IV-15 PREDICTED VARIATION OF CRUISE EFFICIENCY WITH R.P.M. SHOWS SIGNIFICANT INCREASE RESULTING WITH 70% OF HOVER R.P.M.

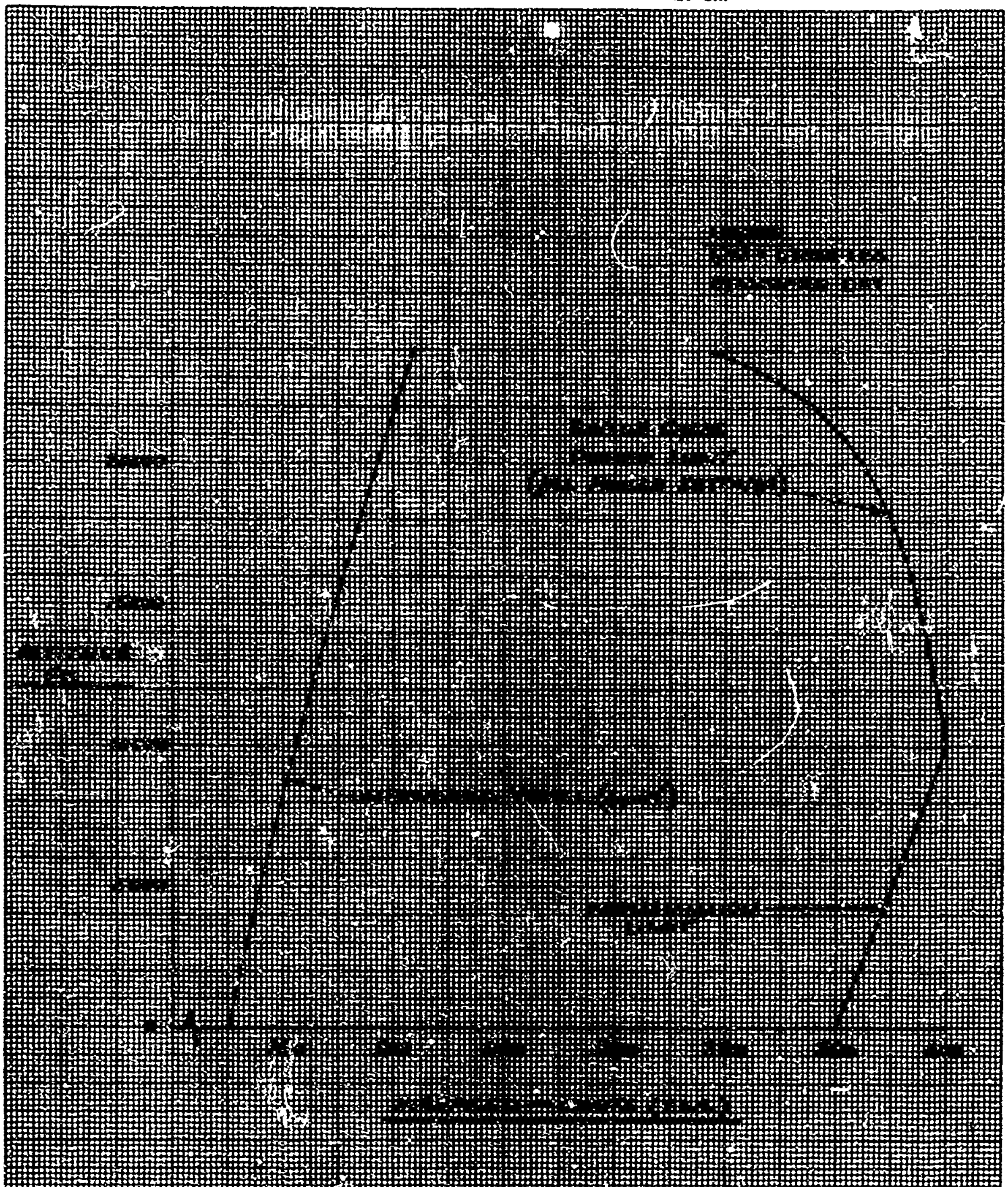

FIGURE IV-16 AIRPLANE MODE FLIGHT ENVELOPE OF TRANSPORT AIRCRAFT WITH ALL ENGINES OPERATING

FIGURE IV-17 FLIGHT ENVELOPE FOR OPERATION WITH ONE OF FOUR ENGINES OUT

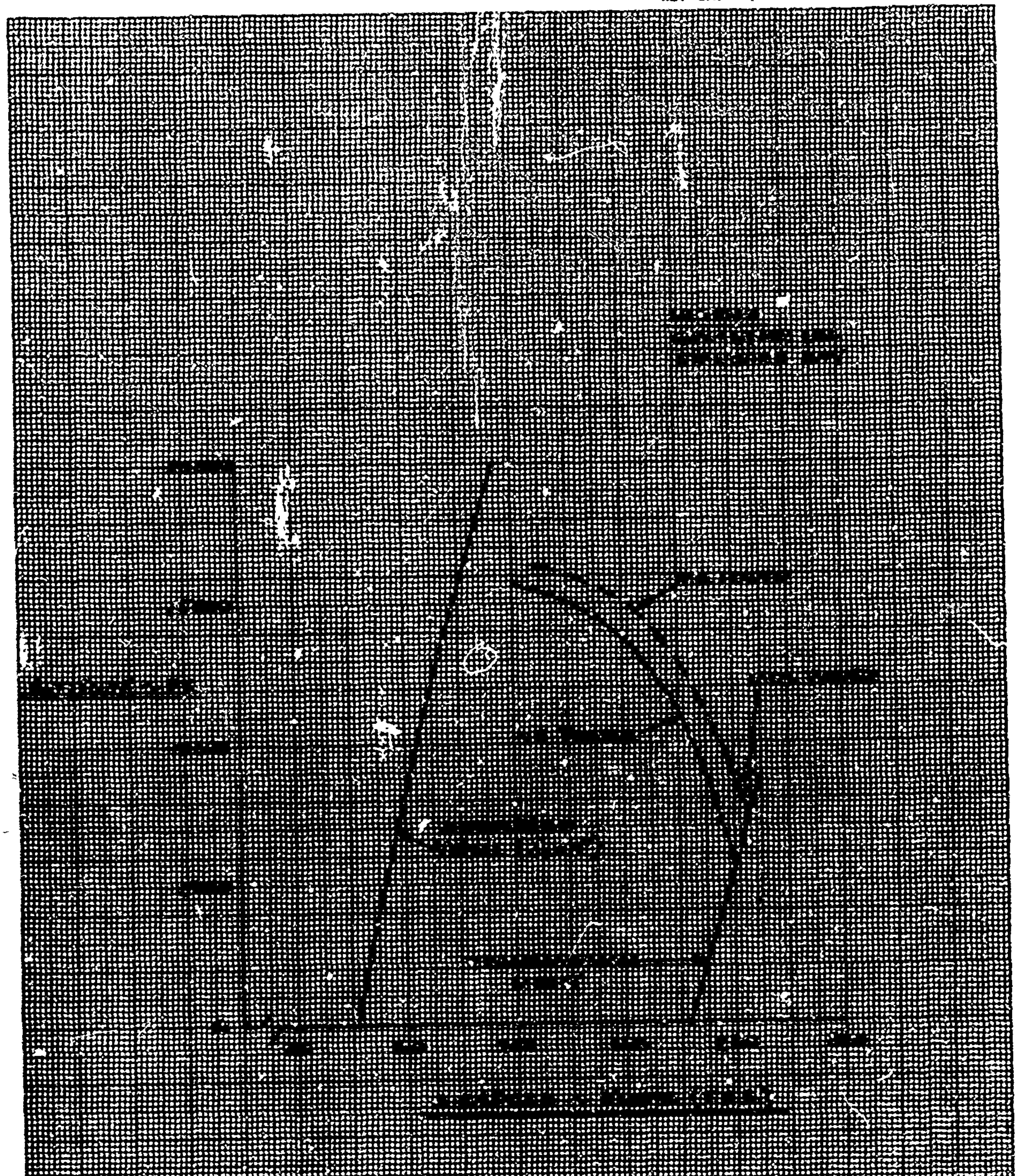

FIGURE IV-18 FLIGHT ENVELOPE FOR OPERATION WITH ONE OF TWO ENGINES OUT and are the result of the primary gas generator rpm limit previously mentioned.

As shown in Figure IV-17, the selection of a four-engine configuration enables the 350 kt, 10,000 ft. and 300 kt. sea level transport mission requirement to be performed at less than MIL power with one engine inoperative. The two-engine aircraft would provide a 275 kt. 10,000 ft. cruise or 263 kt. sea level cruise at MIL power with one engine out.

E. Specific Range

Specific ranges are presented in Figure IV-19 for a range of operational gross weights and altitudes. The maximum endurance data for the 67,000 pound vehicle is also presented.

The ferry range of this aircraft is 26,000 miles with an overload (STOL) take-off gross weight of 74,000 lb.

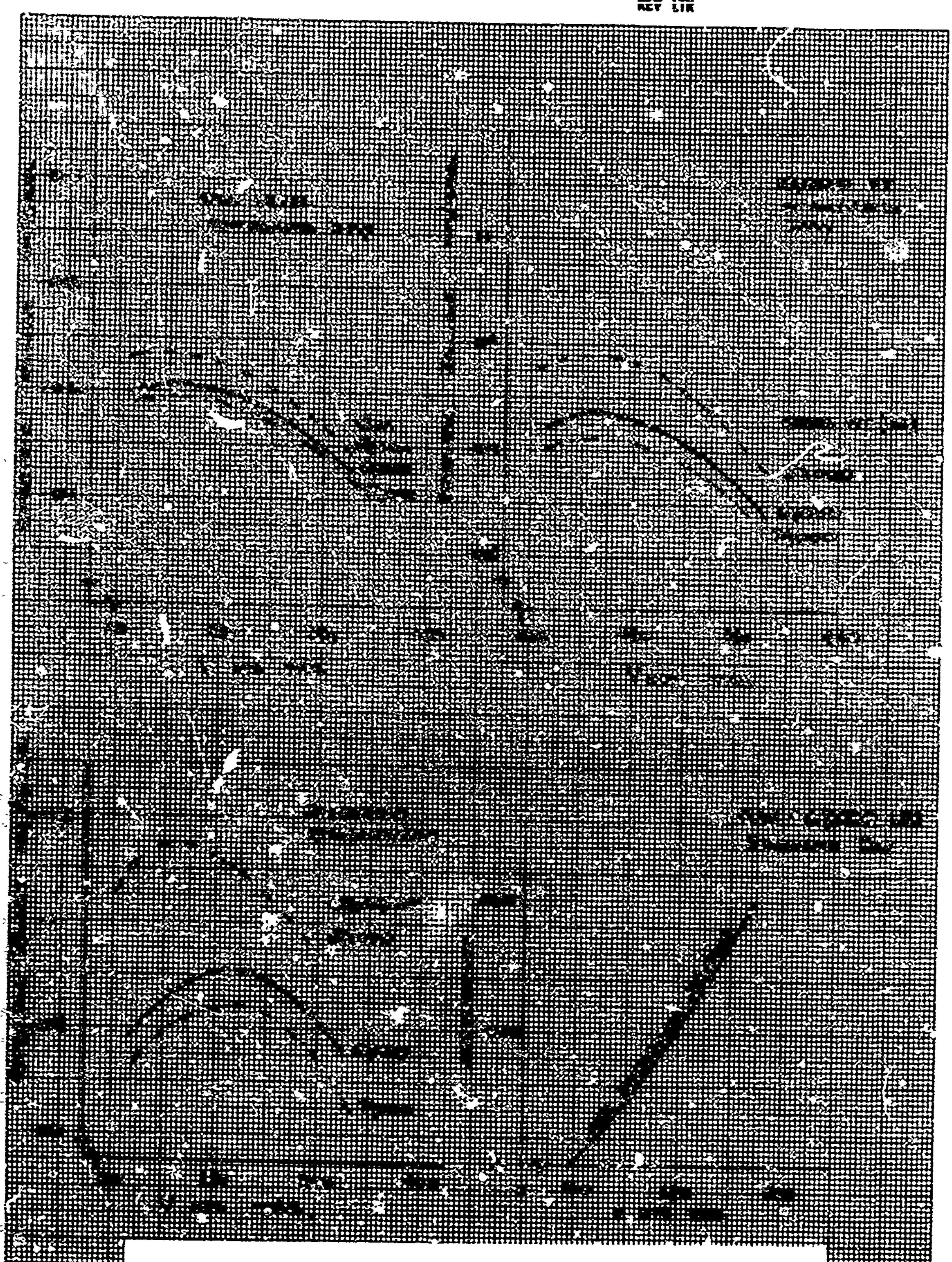

FIGURE IV-19 RANGE PERFORMANCE OF MODEL 215

3. RESCUE MISSION

The rescue mission performance has been considered secondary to the transport mission in so far that no design compromises have been made to accommodate this requirement. The rescue mission described in Section III, is such that the initial and final cruise leg distances are left to be determined.

Two configurations have been considered for the rescue role. First, a converted transport VTOL rescue aircraft with a take-off gross weight of 67,000 lb and secondly, an overloaded converted transport with T.O. gross weight of 74,000 lb. For the latter, a STOL take-off is required although VTOL capability is available at mission mid-point and landing.

The mission data for the 67,000 lb T.O. gross weight aircraft is given in Table IV-2 and shows a range of 642 NM. The take off gross weight of 74,000 lb, corresponding to the overload transport aircraft is capable of 1000 NM. range as indicated in Table IV-3. The possibility of using a smaller fuselage for the rescue aircraft was suggested however in view of the acceptable performance of the overloaded transport. This refinement was considered unnecessary.

A. Overload Gross Weight STOL Take-Off for Rescue Mission

In order to fly the rescue mission, take-off must be made at an overload gross weight of 74,000 lb. Since this is greater than the hover gross weight at 2,000 ft/ANA hot day, a rolling take-off must be made. These results are shown plotted in carpet form in Figure IV-20. The minimum take-off distance over a 50 ft obstacle is about 455 ft at a lift-off speed of 38 fps and nacelle incidence of 75 degrees.

TABLE IV-2 RESCUE MISSION WITH VTOL TAKEOFF

T.O. GW = 67,000 5% Fuel Allowance

SEGMENT	ALT. (FT)	TEMP.	RANGE	MEAN AIRSPEED KT	FUEL USED
Warm up & Taxi (.033)	0	STD	0	0	220
T.O. (VTOL) Hover (.05)	2000	ANA HOT	0	0	310.0
Climb	to 20000	ANA HOT	48.0	240	1382.
Cruise	20,000	ANA HOT	117	360	2345.7
Cruise	3000	ANA HOT	317	350	5978
Climb	to 7000	ANA HOT	321	210	6101
Loiter	7000	ANA HOT	321	100	8091
Hover	6000	ANA HOT	321	0	9206
Pick up 1200 lb	6000	ANA HOT	321	0	9206
Hover	6000	ANA HOT	321	0	10324
Cruise	3000	ANA HOT	521	350	13785
Climb	to 10000	ANA HOT	526	206	13960
Cruise	10,000	ANA HOT	642	400	15963

MISSION FUEL	15963
RESERVE	1596
TOTAL	17559

TABLE IV-3

OVERLOAD CONFIGURATION RESCUE MISSION

STOL TAKE-OFF

T.O. GROSS WEIGHT 78,000 LBS. (5% FUEL ALLOW.)

Segment	Alt. (FT.)	Temp. °F	Range N.M.	Mean Air-speed Kts.	Fuel Used	Spec Range
Warm Up and Taxi	0	Std Day	0	0	220	NA
T.O. STOL	2,000	ANA Hot	0	-	307	NA
Climb	To 20000	ANA Hot	75.0	259	1,706	NA
Cruise	20,000	ANA Hot	300	360	5,066	.0670
Cruise	3,000	ANA Hot	495	350	8,656	.0543
Climb	To 7,000	ANA Hot	500	216	8,817	NA
Loiter	7,000	ANA Hot	500	100.0	11,067	NA
Hover	6,000	ANA Hot	500	0	12,307	NA
Change Payload	6,000	ANA Hot	500	0	12,307	NA
Hover	6,000	ANA Hot	500	0	13,572	NA
Cruise	3,000	ANA Hot	700	350	17,204	.055
Climb	To 10,000	ANA Hot	706.9	210	17,424	NA
Cruise	10,000	ANA Hot	1,000	400	22,534	.0573

Mission Fuel - 22,534
Reserve Fuel - 22,534

TOTAL FUEL -24,787.4

FIGURE IV-20 OVERLOAD GROSS WEIGHT TAKEOFF PERFORMANCE FOR RESCUE MISSION

The large optimum nacelle incidence is primarily due to the fact that the tilt-rotor aircraft derives most of its lift from the rotors. This results in a strong trade-off between lift and longitudinal acceleration as nacelle incidence is varied. At low nacelle incidence, the acceleration is large but a higher speed is required for lift-off. At high incidence, the reverse is true. Since take-off distance increases with lift-off speed (specifically, optimum lift-off speed) but is inversely proportional to the acceleration there will be a minimum in take-off distance at some intermediate nacelle incidence.

The program used for computing take-off performance is based on a two degree-of-freedom trajectory analysis of the take-off. Equations of motion in the horizontal and vertical directions have been formed with the forces on the airframe defined as functions of velocity. The resulting equations thus comprise a set of simultaneous second-order differential equations which can be solved to give time-histories of accelerations, velocities, and distances travelled in the horizontal and vertical directions. The forces on the airframe are computed from the thrust of the rotors and the power-off lift drag characteristics of the aircraft. Inclined disc momentum theory has been used to give the rotor performance. The STOL analysis has four modes of operation: the first simulates a rolling take-off, the second, a helicopter-type take-off, the third simulates an engine failure during a helicopter-type take-off, and the fourth simulates an accelerate-stop maneuver in the helicopter mode. In all of these modes except the accelerate stop mode, the take-off maneuver is assumed to consist of two segments; a ground run or pre-rotation segment, and an air run or past-rotation segment. The ground run is terminated at some rotation, or lift-off speed, entered as an input; or computed, based on some critical speed requirement (such as stall speed or an engine-out climb requirement). In the accelerate-stop mode, the loss of an engine is assumed at the rotation speed and the aircraft is then rotated into a nose-up attitude for deceleration to a stop. During the ground run segment in all modes, the attitude of the aircraft can be limited by fuselage pitch angle or the height of the nose wheel above the ground or both.

4. ALTERNATE TRANSPORT MISSION

The baseline configuration performance has been computed over the alternate transport mission discussed in Section III. The objective of these calculations was to assess the sensitivity of mission radius and mid-point hover time in terms of payload. The general ground rules used in the primary mission calculations have been applied and the assumption made that changes in payload are taken up by fuel (and additional tankage where necessary).

These calculations were also made using VASCOMP II. The results are given in Figure IV-21. It will be noted that the mission radiusfor the five ton payload and two minute mid-point hover time is in excess of 250 N.M. (Primary Mission Radius). This is due to the more favorable specific range obtained at 10,000 ft. altitude since the mission does not call for a sea level dash as does the primary transport mission. The data obtained gives a trade-off of radius to mid-point hover time ratio of 2.899 NM/Min at the design payload of five tons.

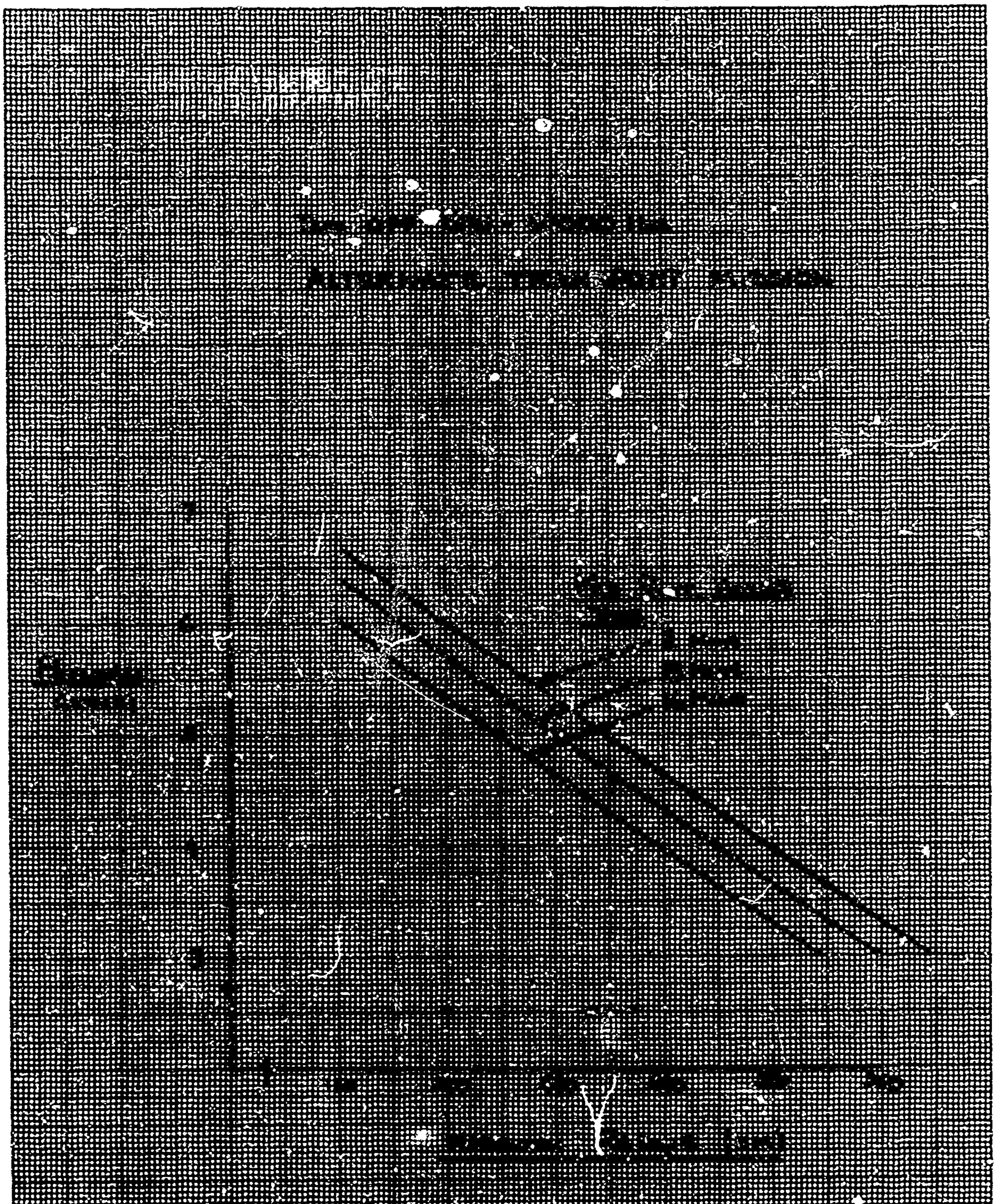

FIGURE IV-21 ALTERNATE TRANSPORT MISSION - PAYLOAD - RADIUS - MID POINT HOVER TIME TRADES

5. HOVER PERFORMANCE METHODOLOGY

In general for VTOL configurations, the hover condition is critical since a deficiency in hover thrust for a particular installed power reduces the payload or mission fuel carried. Further when a matched configuration is considered (equal power for hover and cruise at dash speed), the impact of hovering efficiency on horsepower required and hence on gross weight is large. This situation is not the case of the Model 215 configuration which has a diameter of 55 feet and a dash speed of 400 kt. The impact of hover efficiency is less critical since the power to cruise at 400 knots exceeds the hover power required. The gross weight of the aircraft is still affected by the required maneuver load factor and download since the rotor solidity, and hence the cruise efficiency, is dependent upon these parameters. This section of the report gives the methods used to treat these problem areas and shows correlation of the performance prediction with experiment.

A. Download and Hover Maneuver Load Factor

The wing download technology is based on results of a test program of the Model 160 wing under a CH-47B rotor which was conducted on the Wright-Patterson Air Force Base whirl tower. Simple theoretical methods have been used to extrapolate this data to the present configuration. From simple considerations of swept wing area, uniform inflow theory and the drag of a flat plate normal to a free stream, it is possible to derive an expression for the download thrust to weight ratio as:

$$T/W = 1./\left[1 - \frac{2K^2}{\pi}\left(\frac{C}{D}\right)\left(1 - \frac{C_f}{C}\right)\left(1 - X_c\right) C_{D_V}\right]$$

Where: C/D is the wing chord/diameter ratio
C_f/C is the % chord of the flap
X_c is the nondimensional blade cut out
C_{D_V} is the drag coefficient of a flat plate (Hoerner gives $C_{D_V} = 1.17$)
and K is a constant dependent on the ratio of the induced velocity in the plane of the wing compared with that at infinity.

Deriving K from the 160 tests and calculating the T/W for 30% chord flaps, the result shown in Figure IV-22 is obtained. The use of 15% umbrella flaps as also included in the Model 160 tests provides a 2.6% reduction in T/W. This reduction has been included in the baseline configuration. The umbrella flap provides a reduction in hover download and can also be used as a wing spoiler in low velocity transitional flight to minimize download in transition. This effect will be studied in detail during Phase II as an integral part of wing design.

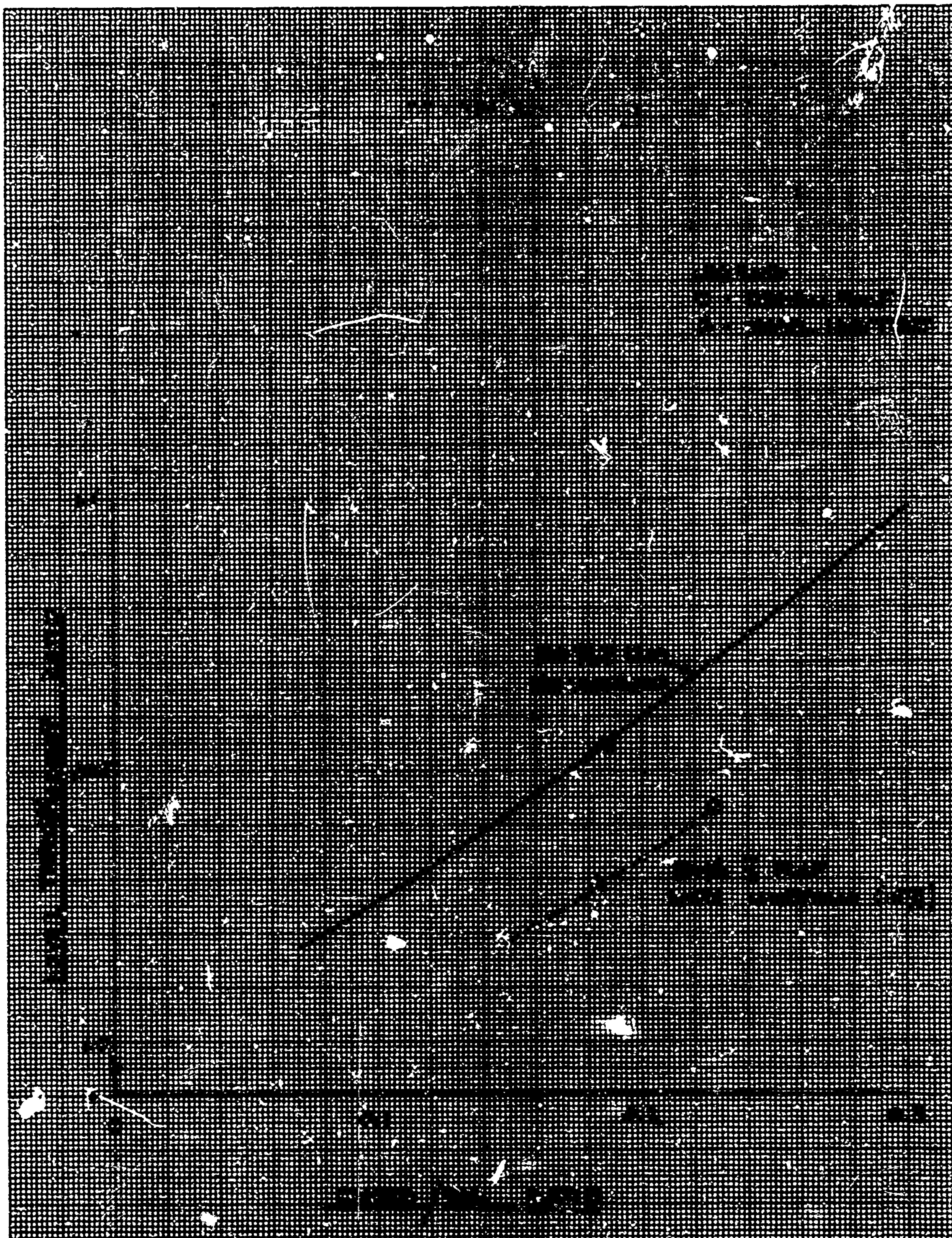

FIGURE IV-22 EXTRA POLATION OF MODEL 160 DOWNLOAD TEST DATA TO MODEL 215 DESIGN

The net thrust maneuver load factor used to size the rotor solidity for the design of the baseline aircraft is 1.15. This 15% margin (in excess of the download T/W of 1.043) is considered to be the service flight envelope rotor limit load factor and is chosen from operational experience with helicopters in both training and combat conditions. In addition, this thrust margin is well in excess of the 5% net thrust margin required by the flying qualities criteria for "level 1" flying qualities. This 1.05 load factor is considered to be the service flight envelope limit load factor for this aircraft.

B. Hover Stall Flutter Margin

The activity factor or solidity of prop/rotors is sized to provide an adequate stall flutter margin. For the USAF Tilt Rotor aircraft (Model 215) the stall flutter margin was defined as the ability to achieve a maneuver load factor of 1.15 and overcome the download at design take-off gross weight of 67,000 lb at an altitude of 2500 ft, 93°. The rotor speed was assumed to be the normal hover value. Since the occurrence of stall flutter is fatigue damaging and does not produce limit rotor loads this stall flutter boundary is assumed to be the limit of the service flight envelope.

Stall flutter is an aeroelastic phenomenon which involves uncoupled blade torsion (twisting) deflections and blade pitch changes due to control system flexibility. The dynamic system consisting of the blade and controls torsional spring, blade pitch inertia; blade structural damping and controls damping is excited by aerodynamic stalling. As the blade stalls at high thrust coefficient the aerodynamic center of the blade moves aft and causes the blade to twist such as to unstall. This phenomenon would not be of such a magnitude as to cause a load problem but as stalling occurs the aerodynamic pitch moment damping becomes negative. With negative damping the twisting due to stall overshoots and rebounds to cause worse stall. This effect oscillates and causes cycles of fatigue loads.

The technology to treat stall flutter has been developed for the helicopter using empirical factors from rotor testing combined with analyses and oscillating airfoil testing. This rotor technology is much more mature than the equivalent propeller technology since the problem has been more limiting for the helicopter. Figure IV-23 illustrates the criteria utilized which relates the rotor

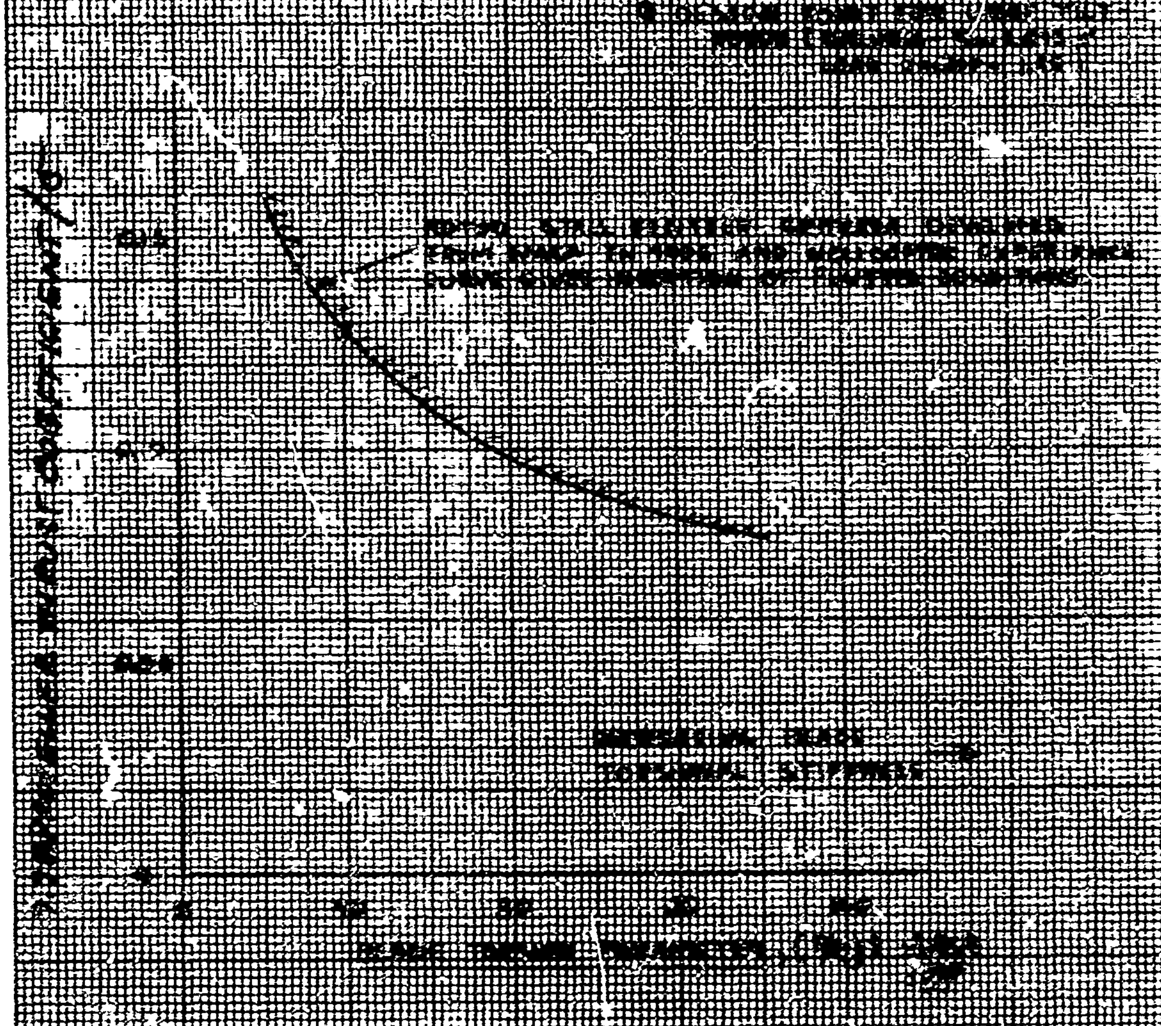

FIGURE IV-23 CRITERIA FOR HOVERING STALL FLUTTER USED TO SIZE BLADE CHORD AND CHECK CONTROL SYSTEM STIFFNESS thrust coefficient to the structural stiffness required of the blade and control system. For the designs discussed in this report a maximum rotor thrust coefficient-solidity ratio of 0.127 was used which required a blade and control system stiffness consistent with contemporary design practice as reflected in the rotor system weight trend curves.

C. Prop/Rotor Hover Performance

Hover performance was computed using the "Explicit Vortex Influence Technique" (EVIT) described in Reference VI-7 and IV-1. This method has provided good correlation with test data in hover for this type of prop/rotor. The Model 160 rotor tests at the Air Force Aero Propulsion Laboratory in February, 1968 reported in Reference IV-2 show this correlation, Figure IV-24. Further examples of methodology substantiation are given in Figures IV-25 to IV-27. This data ranges from the very low disc loading CH-47B rotor to the high disc loading Hamilton Standard propeller test data. In all cases the deviation between data and theory is less than the measurement accuracy of the experimental points indicated by scatter.

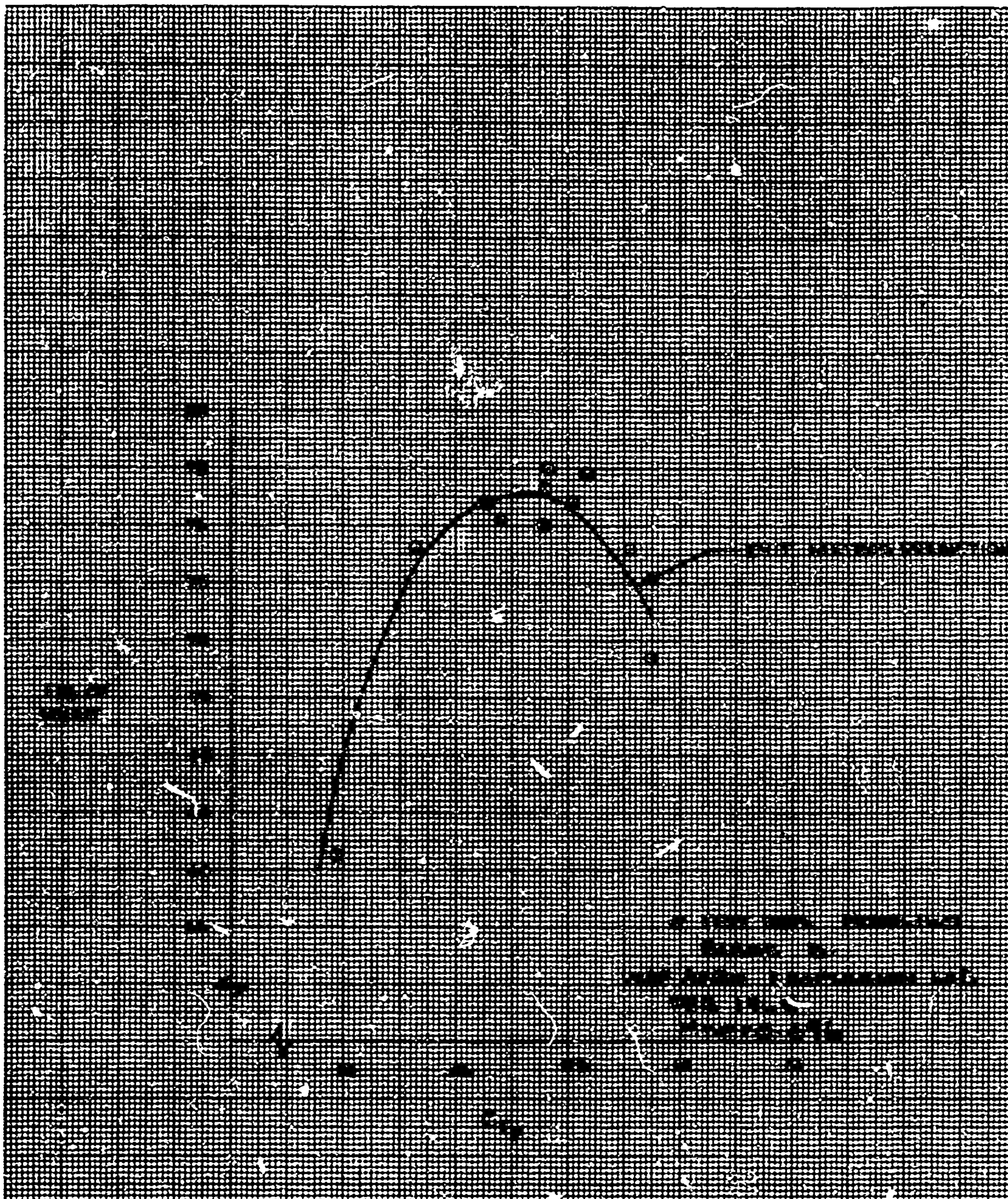

FIGURE IV-24 HOVER PERFORMANCE METHODOLOGY SUBSTANTIATION - Model 160 TESTS

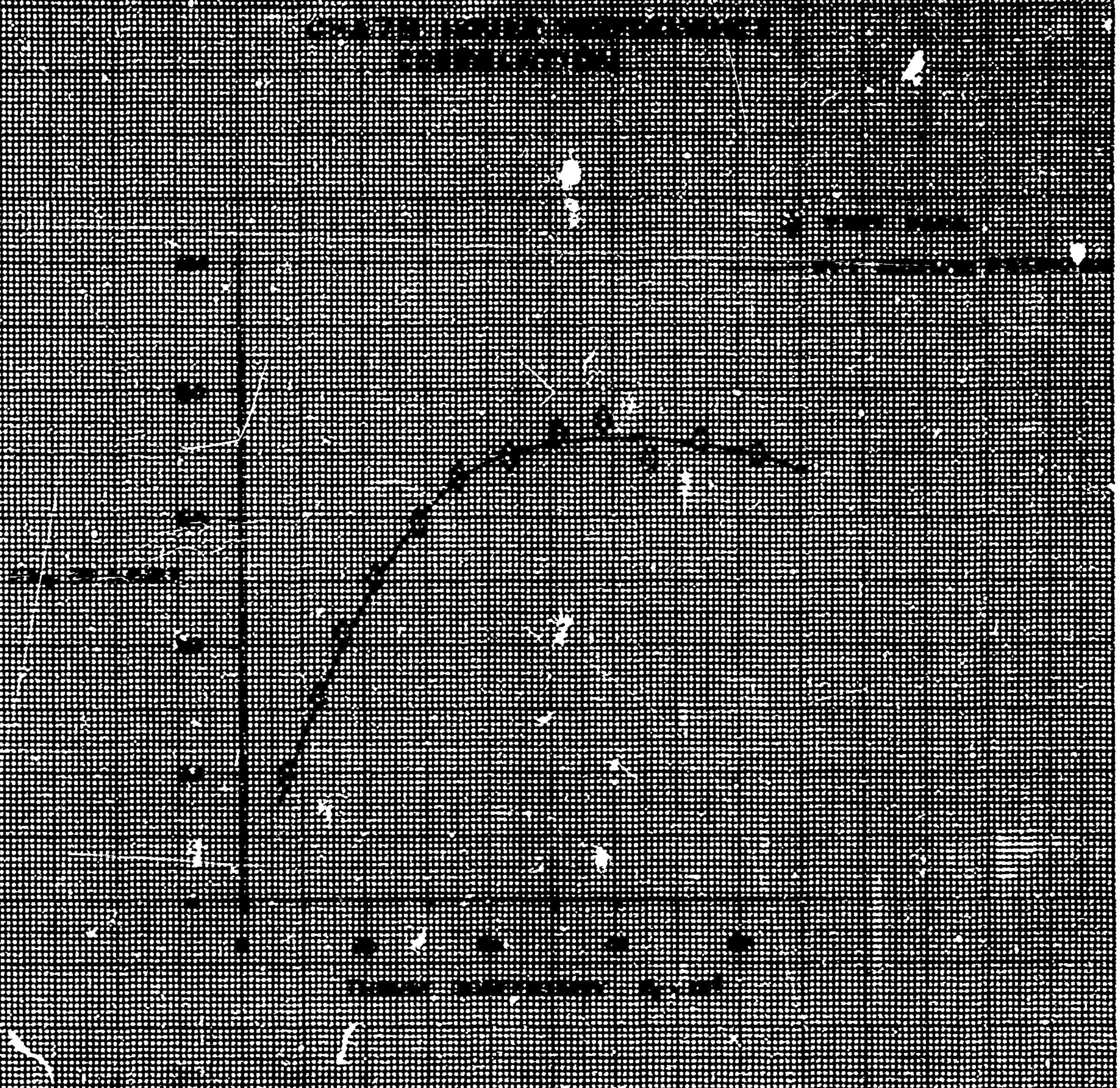

FIGURE IV-25 HOVER PERFORMANCE METHODOLOGY SUBSTANTIATION - CH-47B WHIRL TOWER DATA

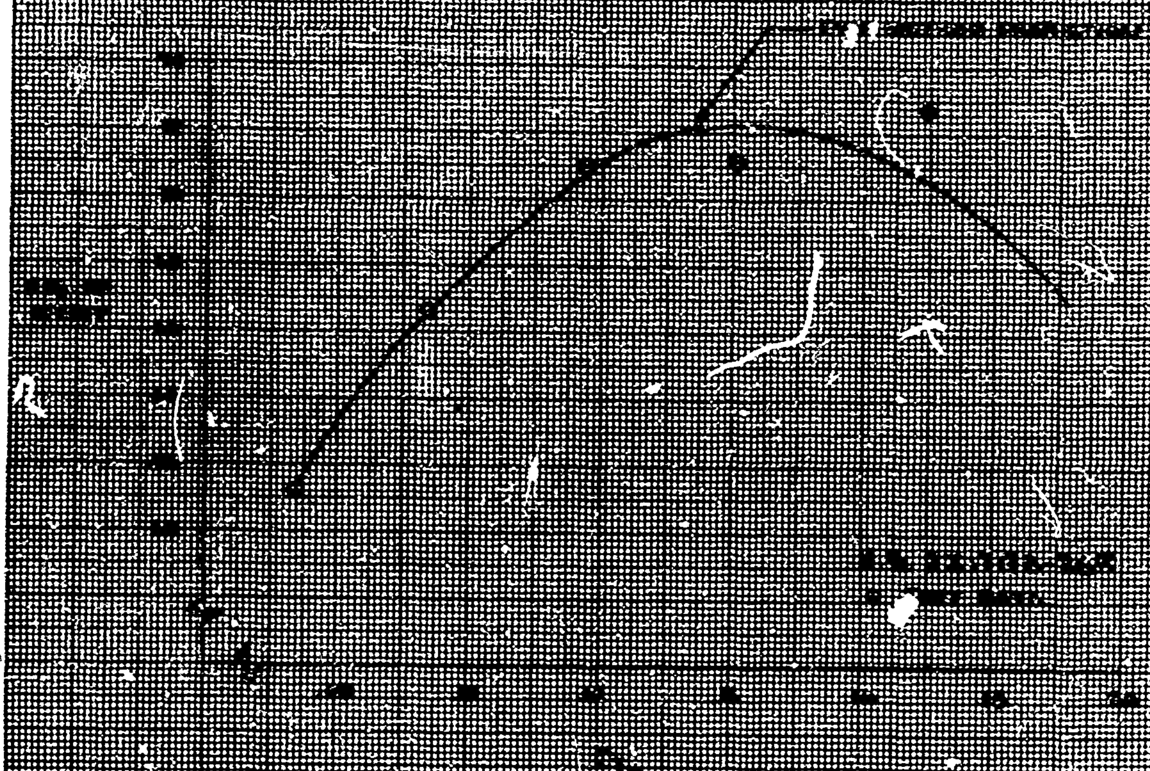

FIGURE IV-26 HOVER PERFORMANCE METHODOLOGY SUBSTANTIATION - H.S. PROP DATA

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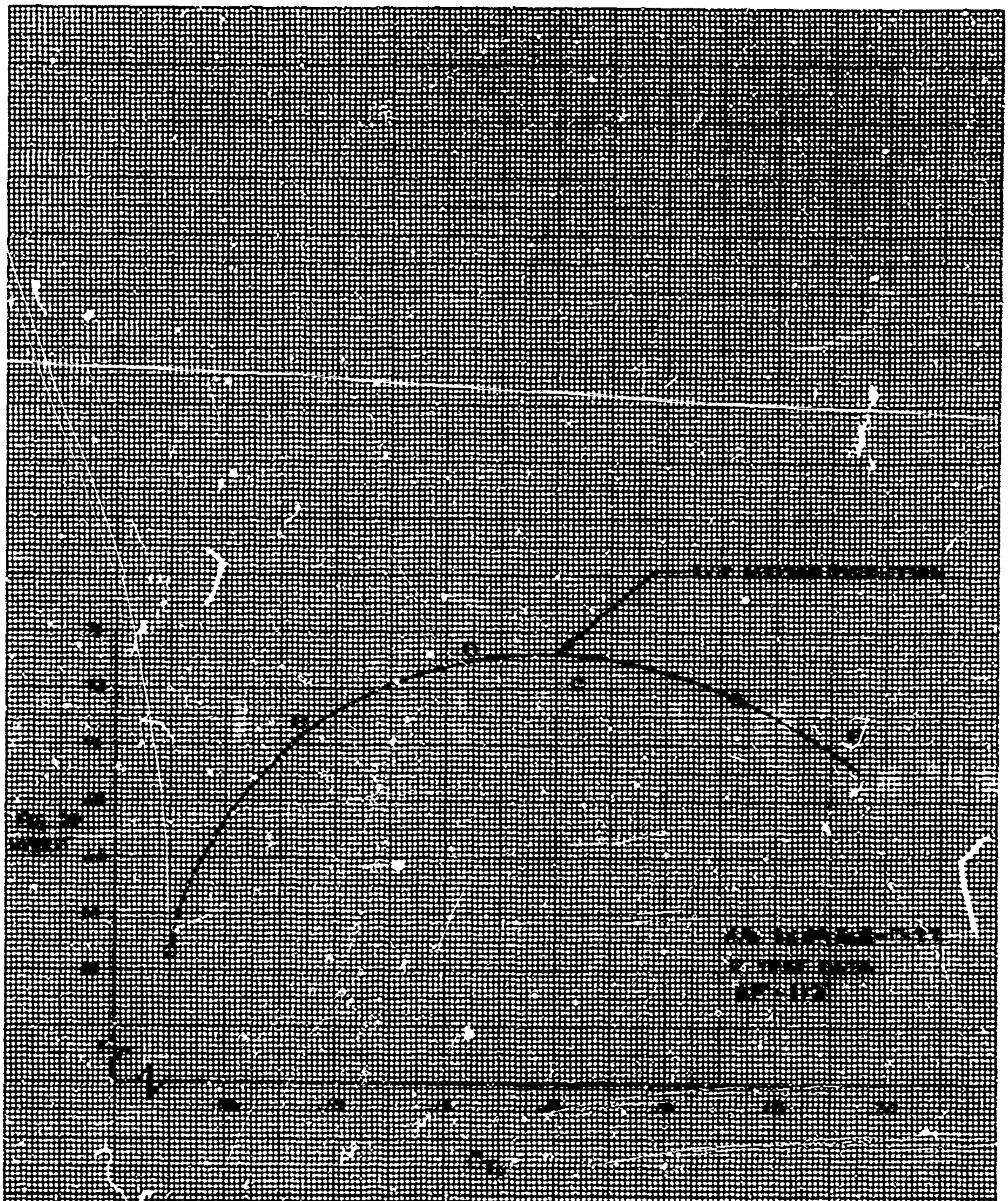

FIGURE IV-27 HOVER PERFORMANCE SUBSTANTIATION - H.S. PROPELLER DATA

The rotor airfoil sections used in this preliminary design study were helicopter blade high Mach Number sections designed at Beoing, Reference IV-3. These low camber sections have the advantage of combining good high Mach number behavior with low pitching moment, an important consideration on rotor blades since the stall flutter tendency is not agravated by these sections.

6. CRUISE PERFORMANCE METHODOLOGY

The methodology available at Boeing-Vertol for the calculation of cruise propeller efficiency consists of two computerized analyses. First, the EVIT program previously mentioned in the hover methodology section which trails the vortex sheet in a regular helix and computes the induced velocity distribution in the plane of the disc. Compressibility effects are included in the airfoil data decks. The second method is the well know Theodorsen technique (also known as the Curtiss-Wright Strip Analysis) where circulation functions are used to determine induced velocities. Both of these methods use airfoil data interpolated from a wide range of sectional data available.

For this study the EVIT program was used to predict cruise prop/rotor performance. Experimental correlations to substantiate the predicted levels of performance using the EVIT analysis are shown in Figures IV-28 and IV-29. In previous studies the Theodorsen technique has been found to be in close agreement with EVIT.

The Curtiss-Wright Strip analysis has been used for many years as a cruise propeller design tool and played an important role in the aerodynamic design of Curtiss propellers such as C130 and Constellation propellers.

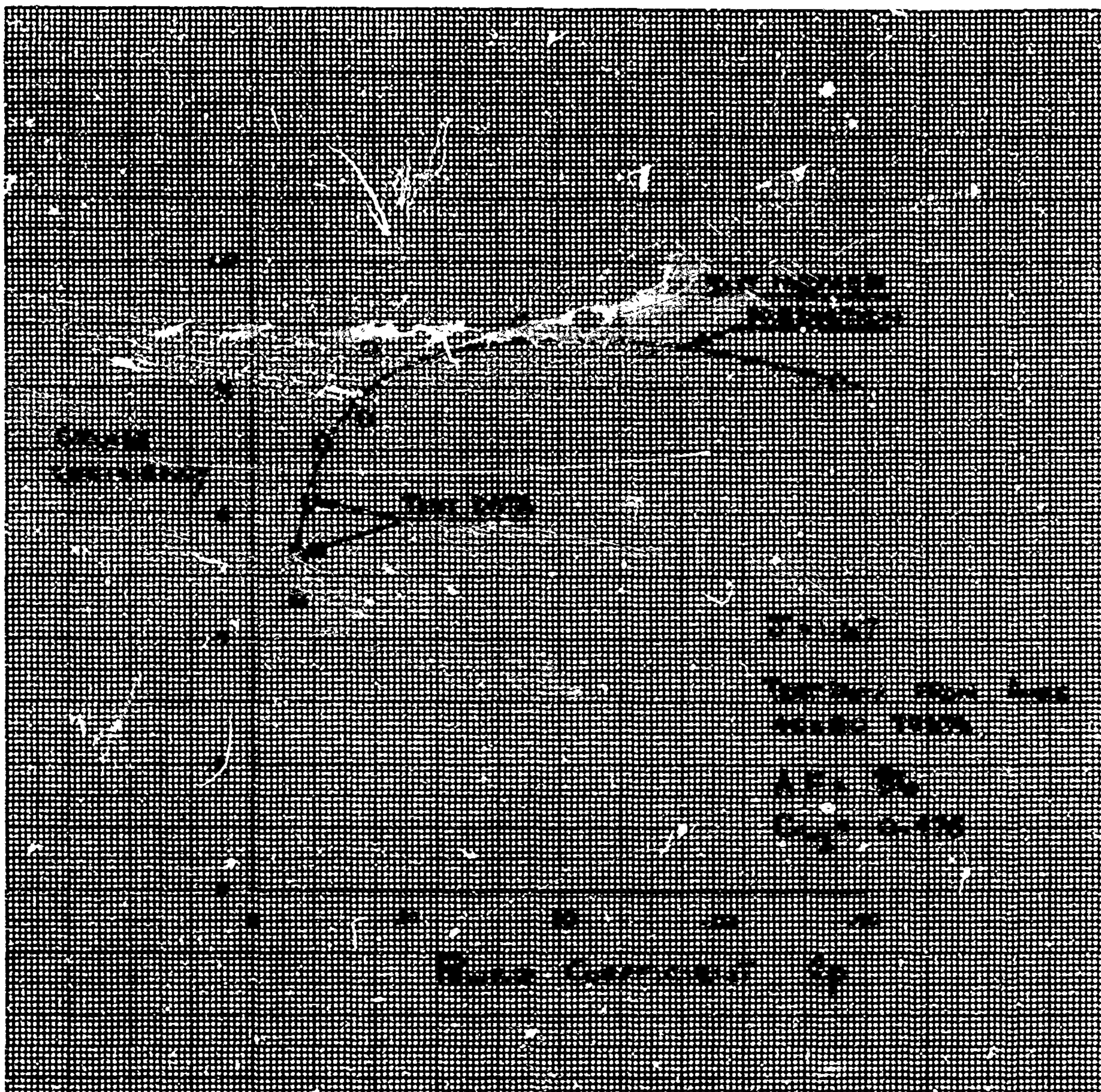

FIGURE IV-28 CRUISE PERFORMANCE METHODOLOGY SUBSTANTIATION - SC142 PROP. H.S. 2FE 16A3 PROPELLER

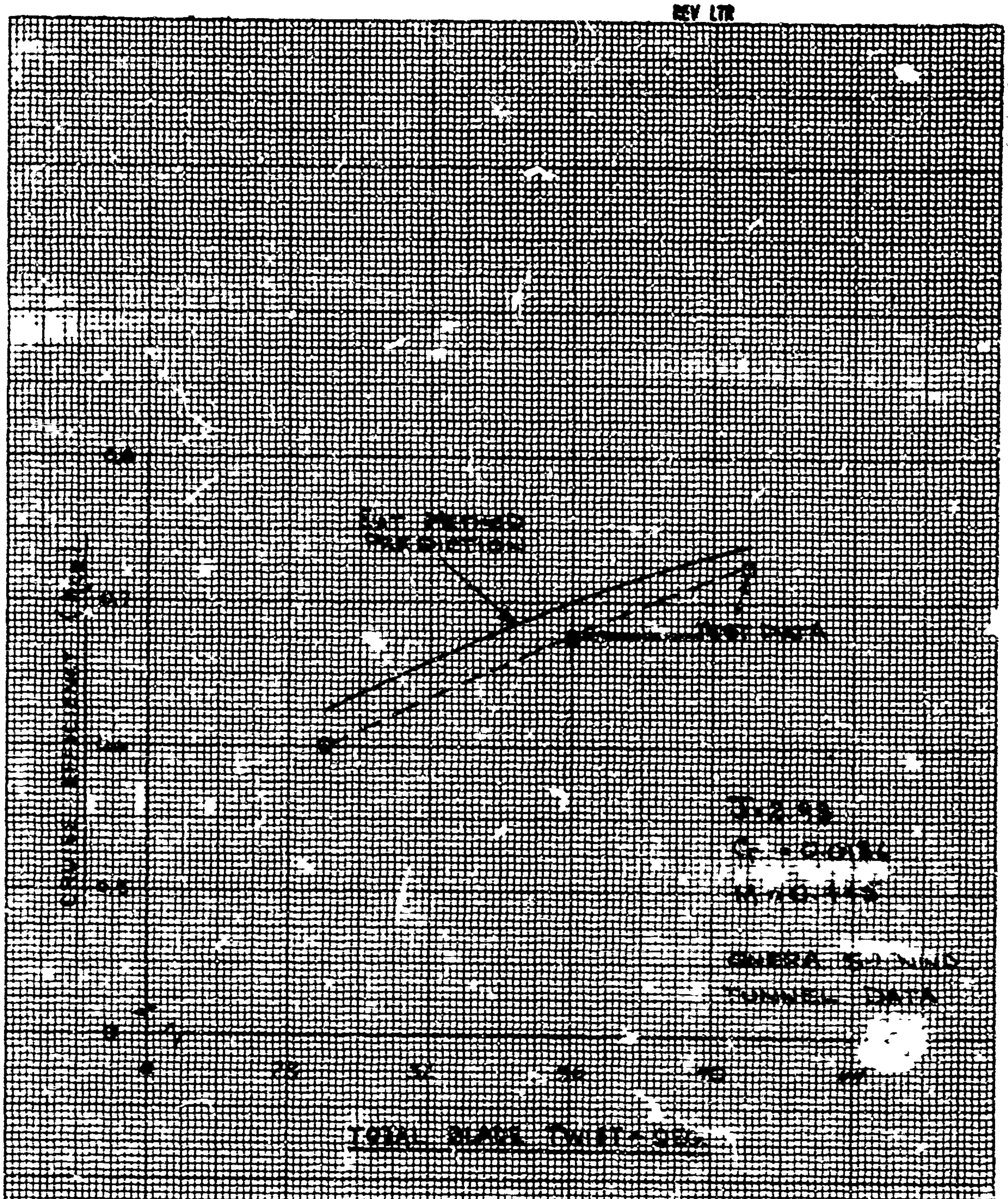

FIGURE IV-29 CRUISE PERFORMANCE METHODOLOGY SUBSTANTIATION - MODEL 160 BLADES AT DESIGN POINT

Test data at high forward flight Mach No. (0.65) obtained in the ONERA S1 Wind Tunnel shows a marked decrease in propulsive efficiency not predicted by either the EVIT program or the Curtiss Wright Strip Analysis. Under these flight conditions the boundary of the propeller wake is determined by the predominant forward flight velocity and hence, the calculation of induced velocity is not likely to be a source of large error. The local profile drag coefficient tables used in the calculations are based on wind tunnel test data and as such reflect the experimental airfoil behavior at high Mach No. In view of the difficulty in understanding the apparent discrepancy an investigation is currently in hand to reevaluate the test data presented since it is knownthat the spinner tares become dominant as tunnel Mach No. increases. The test results of this study will soon be available at which time it is hoped that this problem will be resolved. Since this has not yet resolved, the effect of cruise efficiency on design gross weight is shown in Figure III-19.

7. ENGINE PERFORMANCE METHODOLOGY

The engine cycle data used in sizing aircraft and in the computation of performance is given in Figures IV-30 to IV-33. The data is provided in a "referred" format based on the maximum static sea level horsepower (SHP*).

The cycle data is based on projected 1972 engine technology. The assumptions made in generating this data were as follows.

1. Inlet ram recovery 60%
2. Pressure losses 1.5%
3. Accessory Power 1.0% SHP*

The inlet momentum drag and engine nozzle thrust are included in the available power of the engine. A constant propulsive efficiency of 80% is assummed in converting the thrust/drag to an incremental horsepower. The magnitude of this increment is of the order of ±3% of the engine shaft horsepower available.

The engine limits used were as follows.

$$\frac{N_I}{\sqrt{\theta}\, N_I^*} = 0.982 \quad \text{primary turbine rpm limit}$$

$$\frac{N_{II}}{N_{II}^*} = 1.23 \quad \text{power turbine rpm limit}$$

FIGURE IV-30 ENGINE CYCLE DATA · POWER

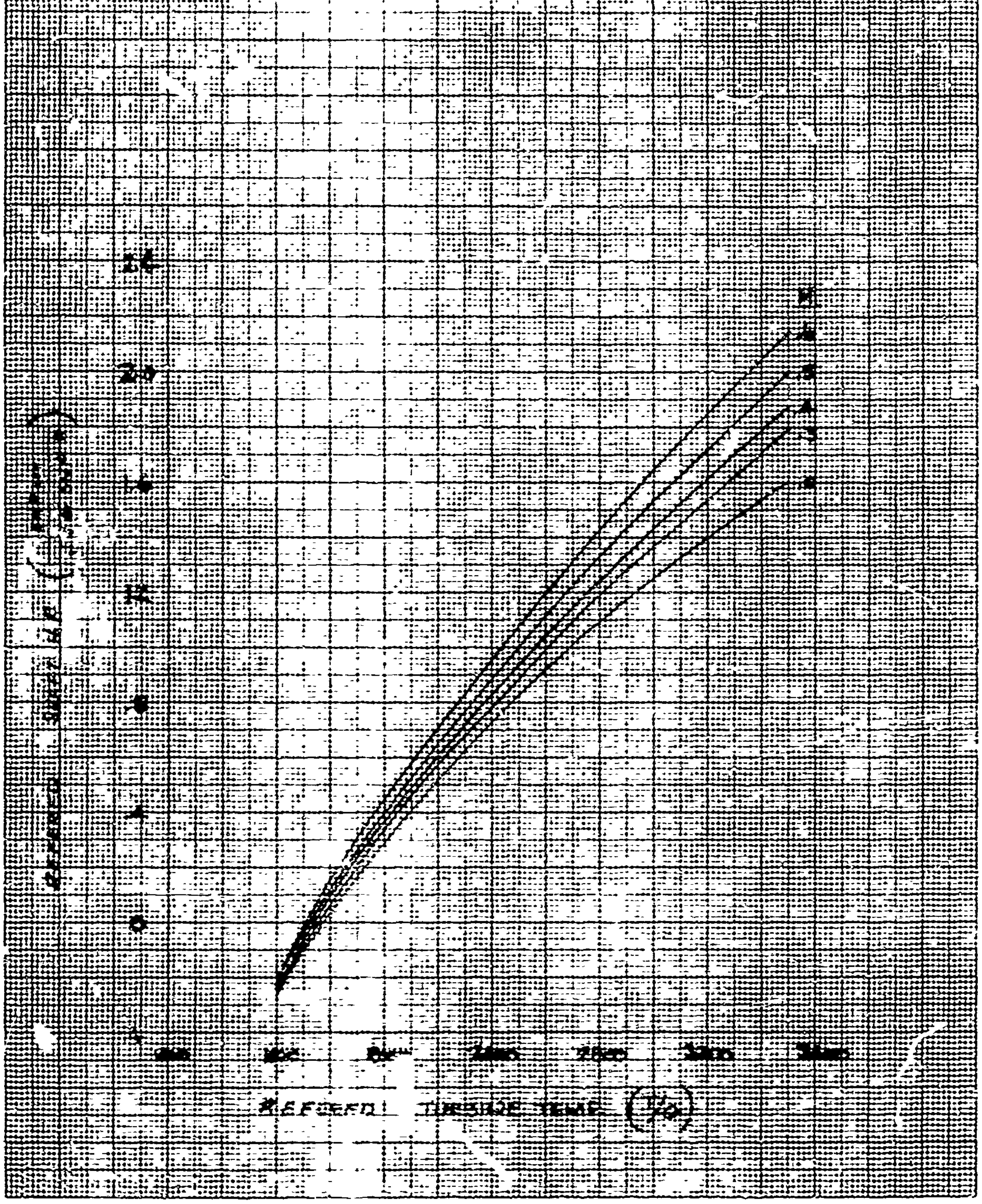

FIGURE IV-31 ENGINE CYCLE DATA - FUEL FLOW

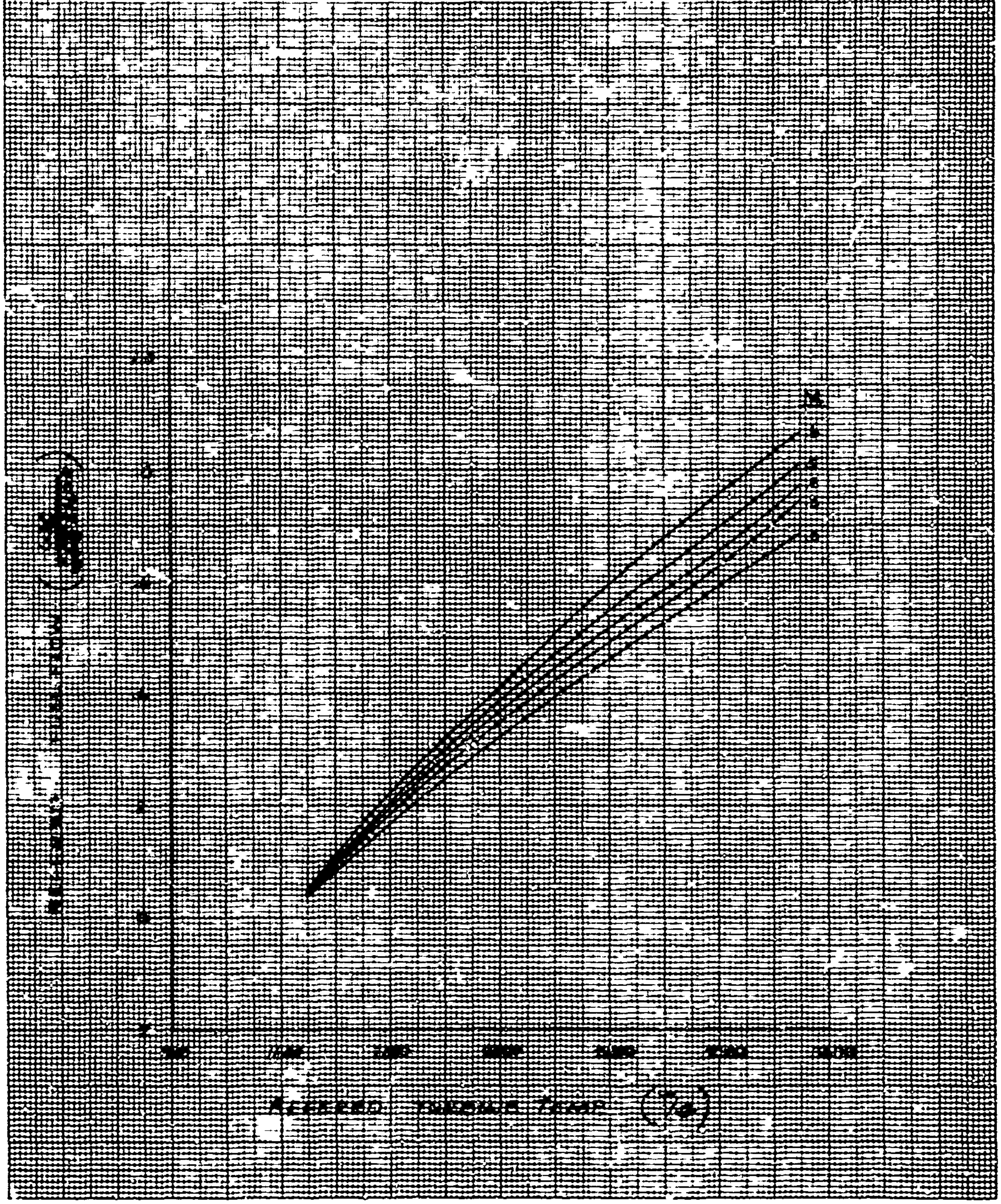

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FIGURE IV-32 ENGINE CYCLE DATA - POWER TURBINE RPM

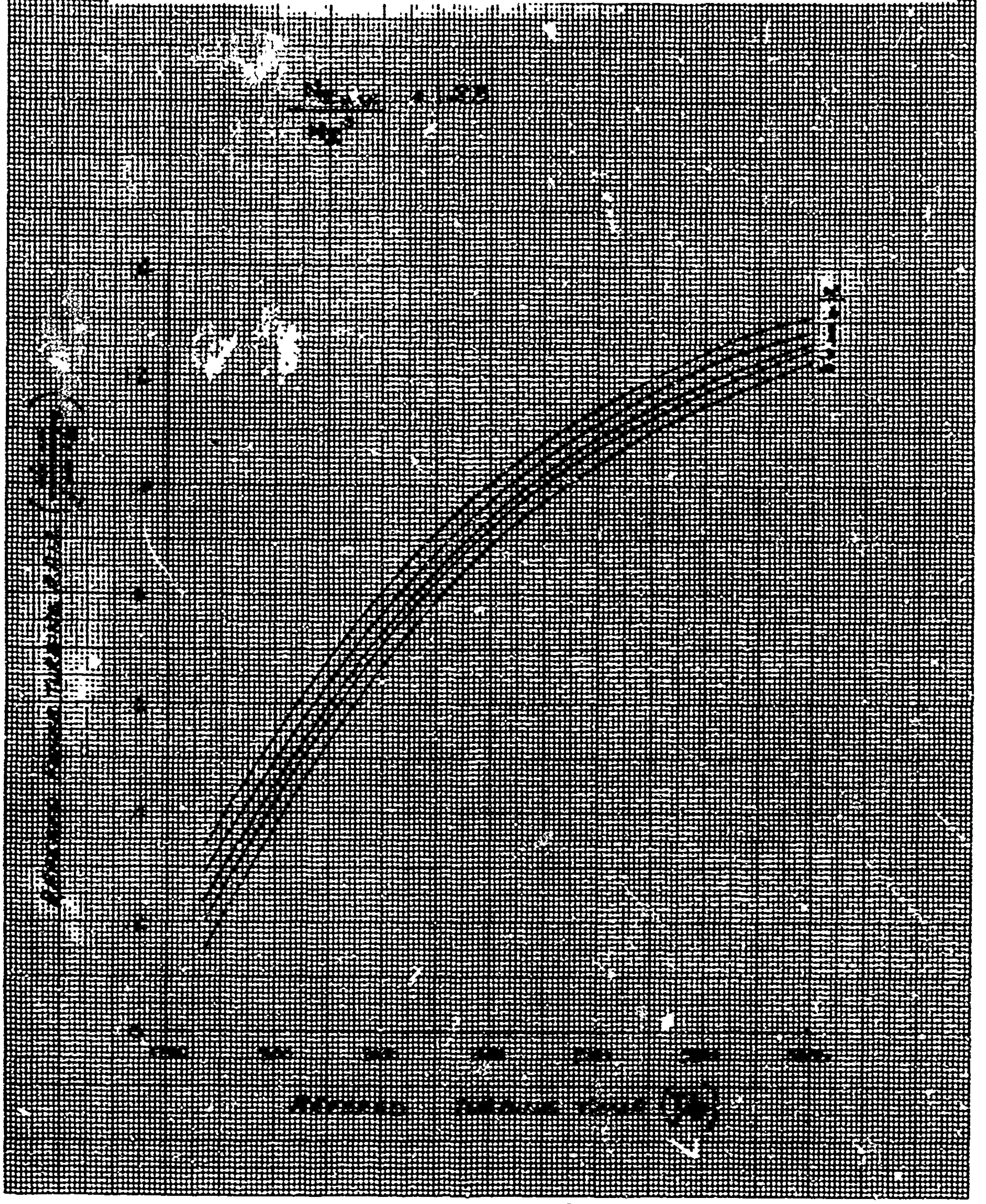
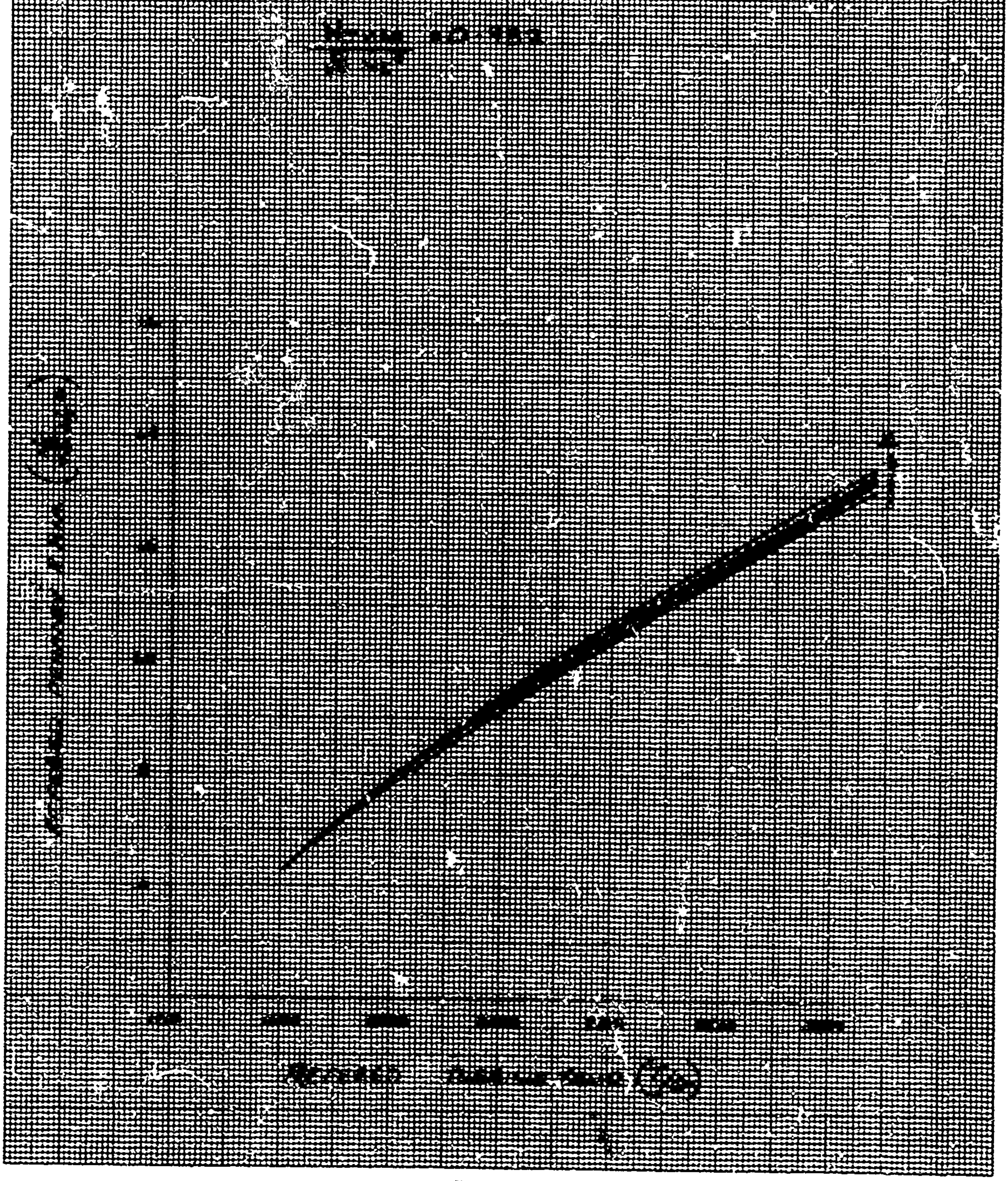


FIGURE IV-33 ENGINE CYCLE DATA - PRIMARY GAS GENERATOR RPM



The turbine temperatures corresponding to the various power settings are.

NRP	T	= 2520°R
MIL	T	= 2565°R
MAX	T	= 2685°R

SECTION V
AIRCRAFT WEIGHT AND BALANCE

1. SUMMARY

The weight of Model 215 was derived in this preliminary design study by using the "V/STOL Aircraft Sizing and Performance Computer Program" (VASCOMP), a program developed for NASA Ames Research Center. This program utilizes the weight estimating methods developed by the Boeing Company, Vertol Division. The weight trends are adjusted for 1972 Technology. Verification of these weights are provided in this section.

During Phase II of this contract, an in-depth system design of the prop/rotor aircraft will be prepared. Particular emphasis will be placed on detail analyses of the wing, engine pod, prop/rotor and associated controls; the weights will be reexamined then.

A summary of the design conditions studied in Phase I is presented in Table V-1. Table V-2 is a group breakdown of the Basic Design Gross Weight. Table V-3 shows the derivation of the Rescue Version of Model 215 from the Basic Model 215 Vehicle.

Center of gravity, payload limitations, balance calculations, moments of inertia and Group Weight Statement, AN-9103-D, toether with a supplement to the "Dimensional and Structural Data" are also presented in this section.

TABLE V-1

SUMMARY OF MODEL 215 DESIGN WEIGHTS

	Weight-Pounds
Weight Empty	45,861
Minimum Flying Weight	47,798
Design Gross Weight	67,000
Maximum Design Gross Weight - STOL	74,000
Landing Gross Weight	68,888
Maximum Overload Gross Weight	
Rescue Gross Weight (642 N. Mi. Range)	67,000
Rescue Gross Weight (1000 N. Mi. Range)	74,000
Ferry Gross Weight (2600 N. Mi. Range)	81,250

TABLE V-2

MODEL 215 WEIGHT BREAKDOWN
BY MAJOR GROUPS FOR BASIC DESIGN
GROSS WEIGHT

	Weight -	Pounds
Wing		4,945
Tail		1,219
Horizontal Tail	667	
Vertical Tail	552	
Body		6,477
Structure	5,463	
Cargo Loading System	980	
Landing Gear		2,546
Flight Controls		5,399
Cockpit	145	
Rotor Upper	2,367	
Rotor Hydraulic	836	
Conventional Aircraft	871	
Tilt Mechanism	1,005	
Stability Augmentation System	175	
Engine Section		1,505
Propulsion		17,856
Engines	2,543	
Air Induction	308	
Exhaust	390	
Lubricating	30	

	Weight -	Pounds
Propulsion (Continued)		
Fuel System	1,636	
Controls	90	
Starting	195	
Prop/Rotor	5,455	
Drive System	7,200	
Auxiliary Power Plant		200
Instruments and Navigation		300
Hydraulics and Pneumatics		335
Electrical		1,248
Electronics		1,093
Armament		50
Furnishings and Equipment		1,812
Accommodations for Personnel	699	
Miscellaneous Equipment	125	
Furnishings	865	
Emergency Equipment	123	
Air Conditioning & Anti-Icing		394
Air Conditioning	255	
Anti-Icing	139	
Auxiliary Gear		24
Contigency		458
Weight Empty		45,861
Crew (3)		645

	Weight	- Pounds
Fuel		10,304
Usable	1C,224	
Unusable	80	
Oil		190
Engine	180	
Trapped	10	
Cargo		10,000
Design Gross Weight		67,000 lbs.

TABLE V - 3

GROSS WEIGHT DERIVATION
FOR RESCUE VERSION OF
MODEL 215

		Weight - Pounds
Design Gross Weight of Transport		67,000
Remove:		
Fuel	-10,224	
Payload	-10,000	
Crew (3)	- 645	
Operating Gross Weight of Transport		46,131
Remove:		
Transport Electronics	-1,093	
Cargo Load System	- 980	
Troop Seats and Prov.	- 434	
Base Weight for Deriving Rescue Version		43,624
Add:		
Crew of (5)		1,075
Electronics		1,572
Communications	224	
Elec. Countermeas	55	
Grd. Fire Detect	14	
Night Operation Equip.	338	
Radio Navigation Aids	184	
Identif. & Beacon	142	
Self-Contained Nav'g.	214	
Terrain Avoidance	256	
Loud Hailer & P.A. Sys.	107	
Shelves and Supts.	38	

Table V-3 (Continued)

		Weight - Pounds
Armament		1,139
Active Defense Prov.	289	
Passive Defense	850	
Ammunition 5.56MM, 6000 RD		220
Guns 5.56MM (2)		70
Mission Equipment		480
Load Handling Gear		105

Rescue Gross WT. Less Fuel and Aux. Tanks 48,285 lbs.

	Design Gross Weight - VTOL Pounds	Overload Gross Weight - STOL Pounds
Gross Weight Less Fuel and Auxiliary Fuel Tank:	48,285	48,285
Basic Fuel	10,224	10,224
Auxillary Fuel	8,016	14,616
Aux. Fuel Tanks	475	875
Rescue Gross Weight	67,000	74,000

2. CENTER OF GRAVITY AND BALANCE CALCULATIONS

The centers of gravity for the various design and alternate gross weights are summarized in Table V-4. Detail balance calculations are included in Tables V-5, 6 and 7.

Studies show that the range between forward and aft center of gravity limits on a typical transport aircraft is five percent of the cargo compartment length. This is equivalent to approximately 15% of MAC for the Model 215 aircraft. To provide a greater loading flexibility the range of allowable center of gravity limits has been increased to 6.7% of cargo compartment length.

The wing has been located so that the Design Gross Weight center of gravity in forward flight is at 25% MAC. The forward flight center of gravity range has been chosen to be from 13% to 33% MAC.

The engine pod pivot point is located at 38% MAC, so that the center line of vertical thrust passes through the center of gravity in the vertical flight condition. The center of gravity range for hover are 30.5% to 45.5% MAC and are limited by prop/rotor blade stresses.

Reference data for the center of gravity calculations are:

1. Horizontal arms are given as fuselage stations.

2. Vertical arms are given as water lines.

3. Fuselage station 0 is 200 inches forward of the forward cargo compartment bulkhead.

4. Water line 0 is 100 inches below the cargo floor.

5. Leading edge of MAC is fuselage station 352.

6. Length of MAC is 153".

7. Engine pod pivot point is: Fuselage station 410, water line 228.

TABLE V-4
MODEL 215 CENTER OF GRAVITY SUMMARY

		Horizontal Flight			Vertical Flight		
	Weight Pounds	% MAC	Fuselage Sta.	Water Line	% MAC	Fuselage Sta.	Water Line
Weight Empty-(Design Gross Wt.)	45,861	21.0	348.2	200.0	40.2	413.3	235.9
Minimum Flying Weight	47,798	19.5	381.8	200.0	37.8	409.8	234.6
Design Gross Weight	67,000	25.2	390.5	193.4	36.2	410.4	218.1
Maximum Design Gross Weight-STOL	74,000	26.7	392.9	187.4	39.4	412.3	212.1
Landing Gross Weight	68,888	26.0	391.8	177.1	38.7	411.2	201.2
Maximum Overload Gross Weight	74,000	26.7	392.9	187.4	39.4	412.3	212.1
Rescue Gross Weight (642 N.Mi. Range)	67,000	21.9	385.5	205.8	34.9	405.4	230.5
Rescue Gross Weight (1000 N.Mi. Range)	74,000	22.3	387.6	197.1	----		----
Ferry Gross Weight (2600 N.Mi. Range)	81,250	28.5	395.6	182.3			

Reference Datums

Fuselage Sta. 0 is 352" forward of leading edge of MAC
Water Line 0 is 100" below cargo compartment floor

Length of MAC - 153"

Center of Gravity Limits: Horizontal Flight 13.0% to 33% MAC
Vertical Flight 30.5% to 45.5% MAC

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TABLE V-5
Model 215 BALANCE CALCULATIONS For Weight Empty and Design Gross Weight

ITEM			STATIONS HORIZONTAL ARM	HORIZONTAL MOMENT	VERTICAL ARM	VERTICAL MOMENT
Rotor Group						
Rotors		5310	298	1,582,380	228	1,210,680
Rotor Spinner		200	284	56,800	228	45,600
Rotor Group	5510		297.5	1,639,180	228	1,256,280
Wing Group	4993		410	2,047,130	228	1,138,404
Tail Group						
Horizontal		714	886	632,604	360	257,040
Vertical		552	817	450,984	286	157,872
Tail Group	1266		855.9	1,083,58[illegible]	327.[illegible]	414,912
Body Group	5518		435	2,400,[illegible]	170	938,060
Alighting Gear						
Nose		514	143	73,502	90	46,260
Main		2,057	555	1,141,635	90	185,130
Alighting Gear	2571		472.6	1,215,137	90	231,390
Flight Controls						
Cockpit		147	140	20,580	55	8,055
Flight		380	440	167,200	160	60,800
SAS		177	190	33,630	65	11,505
Upper		2,389	305	728,645	228	544,692
Hydraulics		845	309	261,105	228	192,660
Tilt Mech.		1,015	410	416,150	238	241,570
Contls In Wing		500	419	209,500	225	
Flight Controls	5453		336.8	1,836,810	214.9	1,171,[illegible]
Engine Section	1520		353	536,560	223	338,96[illegible]
Engines	2568		418	1,073,424	206	529,008
Engine Installation	1028		423	434,844	206	211,768
Fuel System	1652		408	674,016	228	376,656
Drive System						
Transmissions		7000	339	2,373,000	217	1,519,00[illegible]
Drive System-Wing		282	410	115,620	228	64,29[illegible]
Drive System	7282		341.8	2,488,620	217.4	1,583,29[illegible]
Aux. Power Plant	203		514	104,342	208	42,22[illegible]
Instrumentation	306		130	39,780	163	49,87[illegible]

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TABLE V-5 (Continued)
Model 215 BALANCE CALCULATIONS For
Weight Empty and Design Gross Weight

ITEM	WEIGHT	STATIONS — HORIZONTAL ARM	HORIZONTAL MOMENT	VERTICAL ARM	VERTICAL MOMENT
Hydraulics & Pneu.					
Body	131	685		153	
Eng. Pods	80	420		208	
Wing	130	415		231	
Hydraulics & Pneu.	341		177,275		66,713
Electrical					
Body	850	245	208,250	157	133,450
Eng. Pods	180	418	75,240	203	36,540
Wing	232	398	92,336	225	52,200
Electrical	1,262		375,826		222,190
Electronics	1,107	185	204,795	155	171,585
Armament	50	149	7,419	146	7,300
Furnish. & Equipment	1,822	365	665,030	129	235,038
Air Cond. & De-Ice					
Air Cond-Body	260	285	74,100	159	
De-Ice-Body	83	840	69,720	316	
De-Ice-Wing	59	354	20,886	225	
Air Cond. & De-Ice	402		164,706		80,843
Auxiliary Gear	26	579	15,074	133	3,458
Cargo Loading	981	442	433,602	101	99,081
Weight Empty	45,861	384.21	17,620,519	200	9,168,820
Fixed Useful Load					
Crew	645	160	103,200	160	103,200
Trap Liq.	90	409	36,819	228	20,520
Eng. Oil	180	410	73,800	208	37,440
Fixed Useful Load	915		213,810		161,160
Fuel	10,224	408	4,171,392	228	2,331,072
Cargo	10,000	416	4,160,000	130	1,300,000
Gross Weight-Horizontal	67,000	390.53	26,165,712	193.4	12,961,052
Δ Mom-Vert.			1,334,959		1,655,399
Gross Weight - Vertical	67,000	410.46	27,500,661	218.2	14,616,451

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TABLE V-6
Model 215 BALANCE CALCULATIONS For Alternate Gross Weight Conditions

ITEM	WEIGHT	STATIONS			
		HORIZONTAL		VERTICAL	
		ARM	MOMENT	ARM	MOMENT
Weight Empty-Horiz. Flight	45,861	384.2	17,620,576	200.0	9,165,944
Crew	645	160.0	103,200	160	103,200
Trap Liq.	90	409	36,810	228	20,520
Oil	180	410	73,800	208	37,440
10% Fuel	1,022	408	417,139	228	233,107
Min. Flying WT-Horiz. Flight	47,798	381.8	18,251,525	200.0	9,560,201
Δ Moment			1,334,959		1,655,399
Min. Flying WT-Vert. Flight	47,798	409.8	19,586,484	234.6	11,215,600
Design G.W. -Horiz. Flight	67,000	390.5	26,165,778	193.4	12,958,176
+7000# Payload	7000	416	2,912,000	130	910,000
Max. D.G.W. - Horiz. Flight	74,000	392.9	29,077,778	187.4	13,868,176
Max. D.G.W. - Horiz. Flight	74,000		29,077,778		13,868,176
Less 50% Fuel	-5,112	408	-2,085,696	228	-1,665,536
Landing Gross WT-Horiz. Flight	68,888	391.8	26,992,082	177.1	12,202,640
Δ Moment			1,334,959		1,655,399
Landing Gross WT-Vert. Flight	68,888	411.2	28,327,041	201.2	13,858,039
D.G.W. Transport	67,000		26,165,621		12,961,052
Less:					
Fuel	-10224		-4,171,392		-2,331,072
Payload	-10000		-4,160,000		-1,300,000
Crew (3)	-645		- 103,200		-103,200
Electronics	-1093		- 196,740		-174,880
Cargo Load Sys.	- 980		- 426,300		- 98,000
Troop Seats&Prov.	-434		- 156,240		-56,420
Add:					
Electronics	1572		282,960		251,520
Crew (5)	1075		225,750		161,250
Armament	1139		367,897		193,630
Load Hand Gear	105		45,675		10,500
Mission Equipment	480		201,600		75,360
Fuel-Basic	10224		4,171,392	228	2,331,072
Fuel-Add in Wing	7176	408	2,927,808	228	1,636,128
Tank in Wing	430	408	175,440	228	98,040
Fuel-Aux.	840	408	342,720	114	95,760
Tank - Aux.	45	408	18,360	114	5,130
Guns & Ammo	290	400	116,000	120	34,800

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TABLE V-6 (Continued)
Model 215 BALANCE CALCULATIONS For
Alternate Gross Weight Conditions

ITEM	WEIGHT	STATIONS			
		HORIZONTAL		VERTICAL	
		ARM	MOMENT	ARM	MOMENT
Rescue 67,000 Thrust Horiz	67,000	385.48	25,827,351	205.83	13,790,670
Δ Moment			1,334,959		1,655,399
Rescue 67,000 Thrust Vert.	67,000	405.41	27,162,310	230.54	15,446,069
Rescue 67,000 D.G.W.	67,000		25,827,351		13,790,670
Add:					
Δ Fuel Aux.	6600	408	2,692,800	114	752,400
Δ Tank Aux.	400	408	163,200	114	45,600
Rescue 74,000 Thrust Horiz	74,000	387.61	28,683,351	197.14	14,588,670

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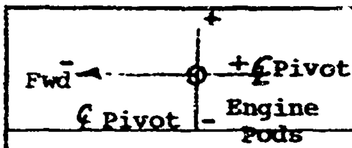

TABLE V - 7

BALANCE CALCULATIONS
For About Pivot Point

Pivot @ Fuselage Sta. 410.
Waterline 228

ITEM	WEIGHT	STATIONS			
		HORIZONTAL		VERTICAL	
		ARM	MOMENT	ARM	MOMENT
Thrust Line Horizontal	(22,194)	(-67.4)	(-1,495,179)	-7.3	(160,220)
Rotors	5,310	-112	-594,720	0	0
Transmissions	7,000	-71	-497,000	-11	-77,000
Engines	2,568	+8	+ 20,544	-22	-56,496
Engine Installation	1,027	+13	+13,351	-22	-22,594
Rotor Spinner	200	-126	-25,200	0	0
Nacelles	1,520	-57	-86,640	-5	-7,600
Upper Controls	2,389	-101	-241,289	0	0
Hydraulics	845	-101	- 85,345	0	0
Tilt Mechanism	1,015	0	0	+10	+10,150
Engine Oil	180	0	0	-20	-3,600
Hydraulics	60	+8	+ 480	-22	-1,320
Electrical	80	+8	+640	-22	-1,760
Thrust Line Vertical	(22,194)	(-7.3)	(-160,220)	(+67.4)	(+1,495,179
Moment Change			+1,334,959		+1,655,399

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3. CARGO CENTER OF GRAVITY LIMITATIONS

In order to maintain a center of gravity within the center of gravity limits, payload loading restrictions must be established. The centroid of the payload for a most forward and most aft airplane center of gravity at various payload weights have been calculated and plotted on Figure V-1. Both horizontal and vertical flight have been considered and the composite limitations are shown.

4. SENSITIVITY OF DESIGN GROSS WEIGHT AND DESIGN RANGE TO FIXED EQUIPMENT WEIGHTS

Sensitivity studies conducted for this aircraft apply mainly to performance and sizing and are discussed in Section III of this document. Exceptions to this are Figures V-2A and B.

Figure V-2A demonstrates the effect of weight variation on the basic mission range, maintaining the design gross weight of 67,000 pounds.

Figure V-2B demonstrates the effect of variation of weight on the gross weight, maintaining the basic mission and performance capability.

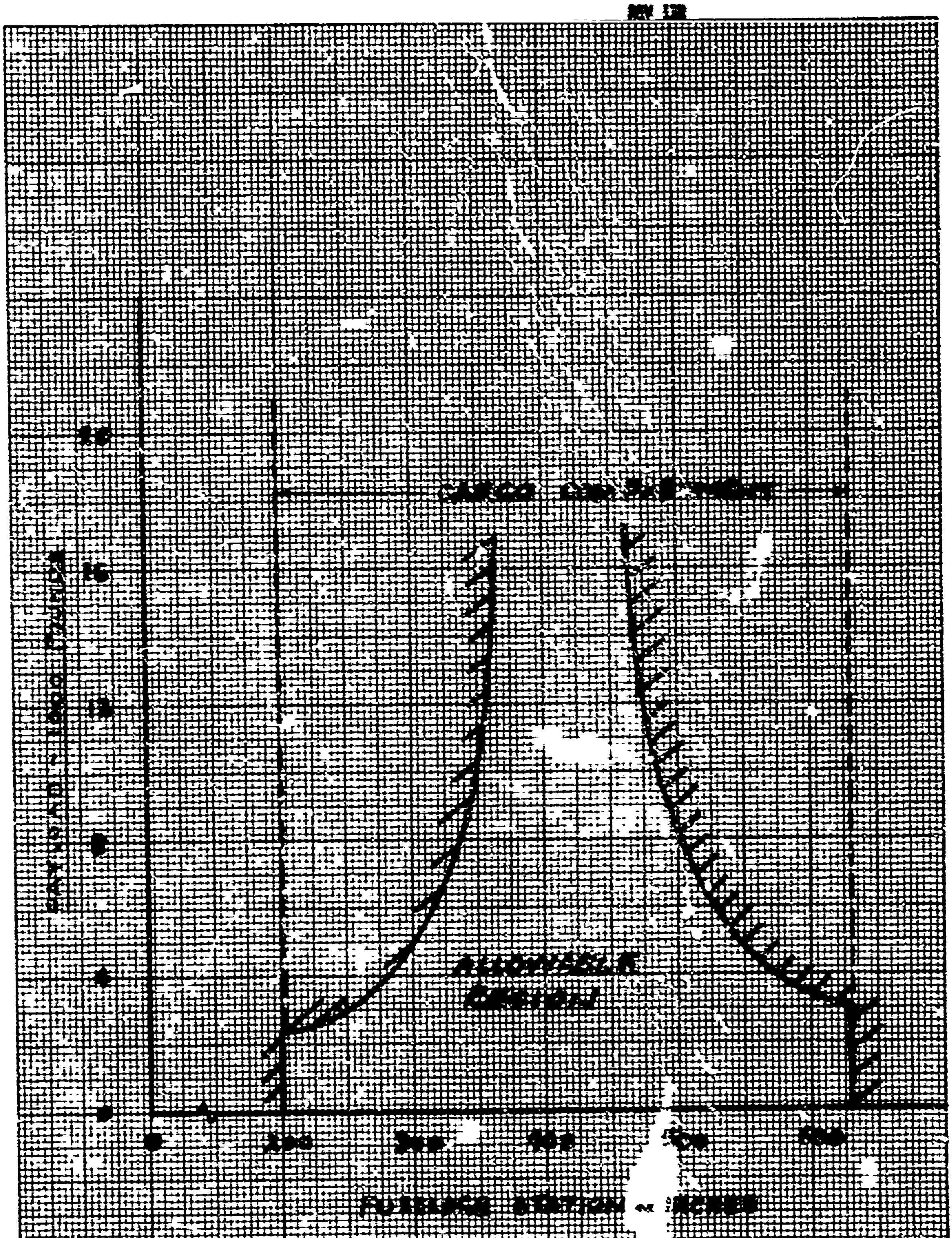
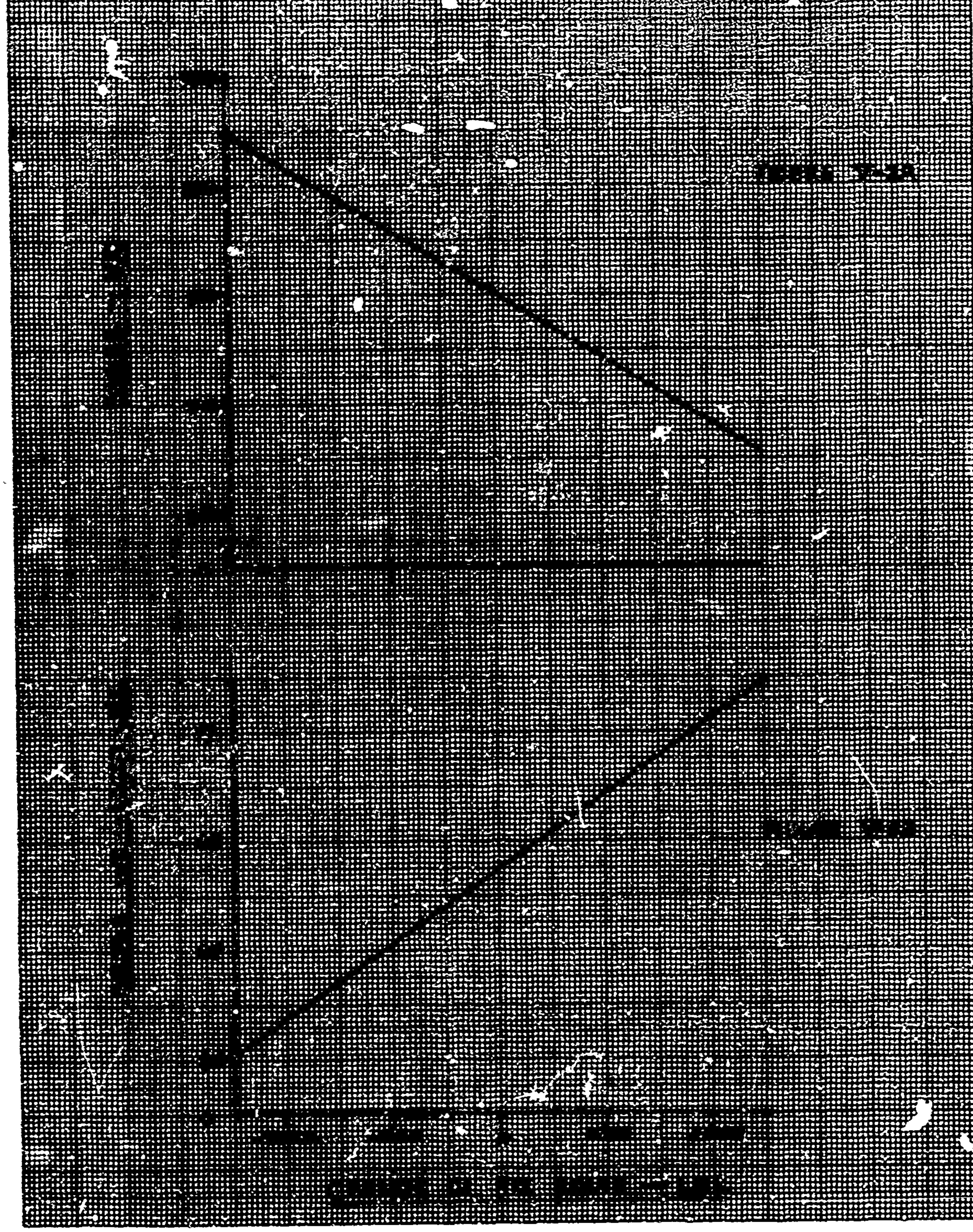


FIGURE V-1   PAY LOAD LOADING LIMITATIONS FOR MODEL 215

FIGURE V-2 EFFECT OF FIXED EQUIPMENT WEIGHT CHANGE ON RANGE AND GROSS WEIGHT FOR MODEL 215



5. MOMENTS OF INERTIA

The Moments of Inertia of the aircraft at the various design gross weight conditions are summarized in Table V-8. The major component moments of inertia are broken down into wing and contents, body and contents, engine pods, fuel and payload in Table V-9 to provide a flexibility for relocation of the components if necessary.

TABLE V - 8

SUMMARY OF MOMENTS OF INERTIA FOR MODEL 215

		Horizontal Flight					Vertical Flight				
	Gross Weight (LBS.)	Center of Gravity		Inertia-Slug Feet^2			Center of Gravity		Inertia-Slug Feet^2		
		Fuselage Sta.	Water Line	Roll	Pitch	Yaw	Fuselage Sta.	Water Line	Roll	Pitch	Yaw
Design Gross Weight	67,000	390.5	193.4	981,304	240,377	1,126,372	410.4	218.1	1,028,560	244,124	1,109,183
Rescue Gross Weight	74,000	392.9	187.4	989,604	250,157	1,143,821	410.9	209.8	1,042,495	259,984	1,131,357
Minimum Flying Weight	47,798	381.8	200.0	958,338	225,313	1,089,382	409.8	234.6	999,689	231,044	1,068,303
Landing Gross Weight	68,888	391.8	184.4	986,169	243,457	1,134,156	411.1	217.4	1,040,768	253,524	1,123,645

TABLE V - 9

COMPONENT MOMENTS OF INERTIA FOR MODEL 215 AT BASIC DESIGN GROSS WEIGHT (SLUG - FEET^2)

		Horizontal Flight		Vertical Flight		Horizontal Flight			Vertical Flight		
	Weight Lbs.	Fuselage Sta.	Water Line	Fuselage Sta.	Water Line	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Wing and Contents	8,006	408.0	227.8	408.0	227.8	72,770	3,058	73,884	72,770	3,058	73,884
Body and Contents	16,576	420.1	157.2	421.1	157.2	19,675	156,420	156,000	19,675	156,675	156,000
Engine Pods and Contents	22,194	342.6	220.7	402.7	295.4	1,032	11,631	11,600	11,600	11,631	1,032
Fuel	10,224	408.0	228.0	408.0	228.0	9,727	413	9,975	9,727	413	9,975
Payload	10,000	416.0	130.0	416.0	130.0	3,166	12,610	14,877	3,166	12,610	14,877
TOTAL AIRCRAFT DESIGN GROSS WEIGHT	67,000	390.5	193.4	410.4	218.1	981,304	240,377	1,126,372	1,028,560	244,214	1,109,183

6. GROUP WEIGHT STATEMENT (AN-9103-D)

A Group Weight Statement, AN-9103-D, is provided. A supplement to the "Dimensional and Structural Data" is included to clarify data used to obtain these weights.

TABLE V - 10

GROUP WEIGHT STATEMENT

ESTIMATED

(Cross out those not applicable)

MODEL 215 PROP/ROTOR TRANSPORT

CONTRACT NO. ____________

AIRPLANE, GOVERNMENT NO. ____________

AIRPLANE, CONTRACTOR NO. ____________

MANUFACTURED BY ____________

		MAIN	AUXILIARY
ENGINE	MANUFACTURED BY		
	MODEL		
	NO.		
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

AN-9103-D
NAME ______________
DATE ______________

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE ______________
MODEL ______________
REPORT ______________

1	WING GROUP		4945
2	CENTER SECTION - BASIC STRUCTURE		
3	INTERMEDIATE PANEL - BASIC STRUCTURE		
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS LBS.)		
5			
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM LBS.)		
7	AILERONS (INCL. BALANCE WEIGHT LBS.)		
8	FLAPS - TRAILING EDGE		
9	- LEADING EDGE		
10	SLATS		
11	SPOILERS		
12	SPEED BRAKES		
13			
14			
15	TAIL GROUP		1,219
16	STABILIZER - BASIC STRUCTURE		
17	FINS - BASIC STRUCTURE (INCL. DORSAL LBS.)		
18	SECONDARY STRUCTURE (STAB. & FINS)		
19	ELEVATOR (INCL. BALANCE WEIGHT LBS.)		
20	RUDDERS (INCL. BALANCE WEIGHT LBS.)		
21	Horizontal	667	
22	Vertical	552	
23	BODY GROUP		6,477
24	FUSELAGE Structure	5,497	
25	BOOMS - BASIC STRUCTURE		
26	SECONDARY STRUCTURE - FUSELAGE OR HULL		
27	- BOOMS		
28	- SPEEDBRAKES		
29	- DOORS, PANELS & MISC.		
30	Cargo Loading System	980	
31	ALIGHTING GEAR GROUP - LAND (TYPE:)		2546
32	LOCATION / WHEELS, BRAKES TIRES, TUBES, AIR / STRUCTURE / CONTROLS		
33			
34	Main	2037	
35	Nose	509	
36			
37			
38			
39			
40	ALIGHTING GEAR GROUP - WATER		---
41	LOCATION / FLOATS / STRUTS / CONTROLS		
42			
43			
44			
45			
46	Flight CONTROLS GROUP		5399
47	COCKPIT CONTROLS	145	
48	AUTOMATIC PILOT (SAS)	175	
49	Rotor	3203	
50	Conventional = 871, Tilt = 1,005	1876	
51	ENGINE SECTION OR NACELLE GROUP		1505
52	INBOARD	---	
53	CENTER	---	
54	OUTBOARD	1355	
55	DOORS, PANELS & MISC.	150	
56			
57	TOTAL (TO BE BROUGHT FORWARD)		22091

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NAME ______________________

DATE ______________________

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE ______________

MODEL ______________

REPORT ______________

		Auxiliary	Main	
1	PROPULSION GROUP			17856
2		AUXILIARY	MAIN	
3	ENGINE INSTALLATION		2543	
4	AFTERBURNERS (IF FURN. SEPARATELY)			
5	ACCESSORY GEAR BOXES & DRIVES			
6	SUPERCHARGERS (FOR TURBO TYPES)			
7	AIR INDUCTION SYSTEM		308	
8	EXHAUST SYSTEM		390	
9	COOLING SYSTEM			
10	LUBRICATING SYSTEM		30	
11	TANKS			
12	COOLING INSTALLATION			
13	DUCTS, PLUMBING, ETC.			
14	FUEL SYSTEM		1636	
15	TANKS - PROTECTED			
16	- UNPROTECTED			
17	PLUMBING, ETC.			
18	WATER INJECTION SYSTEM			
19	ENGINE CONTROLS		90	
20	STARTING SYSTEM		195	
21	PROPELLER INSTALLATION		5455	
22	Drive System		7209	
23				
24	AUXILIARY POWER PLANT GROUP			200
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP			200
26	HYDRAULIC & PNEUMATIC GROUP			335
27				
28				
29	ELECTRICAL GROUP			1248
30				
31				
32	ELECTRONICS GROUP			1093
33	EQUIPMENT		791	
34	INSTALLATION		302	
35				50
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)			1812
37	FURNISHINGS & EQUIPMENT GROUP		699	
38	ACCOMMODATIONS FOR PERSONNEL		125	
39	MISCELLANEOUS EQUIPMENT		865	
40	FURNISHINGS		123	
41	EMERGENCY EQUIPMENT			
42				394
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP		255	
44	AIR CONDITIONING		139	
45	ANTI-ICING			
46				
47	PHOTOGRAPHIC GROUP			
48	AUXILIARY GEAR GROUP			24
49	HANDLING GEAR		24	
50	ARRESTING GEAR			
51	CATAPULTING GEAR			
52	ATO GEAR			
53				
54				
55	Contingency			458
56	TOTAL FROM PG. 2			22091
57	WEIGHT EMPTY			45861

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NAME ________
DATE ________

GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

PAGE ________
MODEL ________
REPORT ________

1	LOAD CONDITION							
2								
3	CREW (NO. 3)				645			
4	PASSENGERS (NO.)							
5	FUEL	Type	Gals.					
6	UNUSABLE				80			
7	INTERNAL				10224			
8								
9								
10	EXTERNAL							
11								
12	BOMB BAY							
13								
14	OIL				190			
15	TRAPPED			10				
16	ENGINE			180				
17								
18	FUEL TANKS (LOCATION)				---			
19	WATER INJECTION FLUID (GALS)				---			
20								
21	BAGGAGE				---			
22	CARGO				10,000			
23								
24	ARMAMENT				---			
25	GUNS (Location)	Fix. or Flex.	Qty.	Cal.	---			
26								
27								
28								
29								
30								
31								
32	AMMUNITION				---			
33								
34								
35								
36								
37								
38								
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)				---			
*40	BOMB OR TORPEDO RACKS							
41								
42								
43								
44								
45					---			
46	EQUIPMENT							
47	PYROTECHNICS							
48	PHOTOGRAPHIC							
49								
*50	OXYGEN							
51								
52	MISCELLANEOUS							
53								
54								
55	USEFUL LOAD				21,139			
56	WEIGHT EMPTY				45,861			
57	GROSS WEIGHT				67,000			

*If not specified as weight empty.

AN-9103-D
NAME ______
DATE ______

GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

PAGE ______
MODEL ______
REPORT ______

1 LENGTH - OVERALL (FT.)			HEIGHT - OVERALL - STATIC (FT.)				
2	Main Floats	Aux. Floats	Booms	Fuse or Hull	Nacelles Inboard	Nacelles Center	Nacelles Outboard
3 LENGTH - MAX. (FT.)				68.3			19.0
4 DEPTH - MAX. (FT.)				11.5			5.7
5 WIDTH - MAX. (FT.)				9.8			5.7
6 WETTED AREA (SQ. FT.)				2280			
*7 FLOAT OR HULL DISPL. - MAX. (LBS.)							
8 FUSELAGE VOLUME (CU. FT.)		PRESSURIZED 0			TOTAL		

	Wing	H. Tail	V. Tail
9	Wing	H. Tail	V. Tail
10 GROSS AREA (SQ. FT.)	838	257	141
11 WEIGHT/GROSS AREA (LBS./SQ. FT.)	5.9	2.7	3.9
12 SPAN (FT.)	65.8	32.0	11.5
13 FOLDED SPAN (FT.)	---	----	---
14			
15 SWEEPBACK - AT 25% CHORD LINE (DEGREES)	0		
16 - AT % CHORD LINE (DEGREES)	0		
**17 THEORETICAL ROOT CHORD - LENGTH (INCHES)	153	113	193
18 - MAX. THICKNESS (INCHES)	32	17.2	30.1
***19 CHORD AT PLANFORM BREAK - LENGTH (INCHES)	153		
20 - MAX. THICKNESS (INCHES)	32		
***21 THEORETICAL TIP CHORD - LENGTH (INCHES)	153	80.4	115.8
22 - MAX. THICKNESS (INCHES)	32	12.0	17.4

23 DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.)

24 TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.) 38.8

25 AREAS (SQ. FT.) Flaps	L.E.	T.E.	
26 Lateral Controls	Slots	Spoilers	Ailerons
27 Speed Brakes	Wing	Fuse. or Hull	
28			
29			

(LOCATION)	Main	Nose		
30 ALIGHTING GEAR	Main	Nose		
31 LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)	23	40		
32 OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)	8.4	8.4		
33 FLOAT OR SKI STRUT LENGTH (INCHES)	---	---		

34 ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES) ---

35 HYDRAULIC SYSTEM CAPACITY (GALS.)

36 FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
37 Fuel - Internal	Wing	12	1,575	---	---
38	Fuse. or Hull				
39 - External					
40 - Bomb Bay					
41					
42 Oil					
43					

44

45 STRUCTURAL DATA - CONDITION	Fuel in Wings (Lbs.)	Stress Gross Wt.	Ult. L.F.
46 FLIGHT	10,224	67,0[illegible]	4.5
47 LANDING	5,112	68,888	3.5
48			
49 MAX. GROSS WEIGHT WITH ZERO WING FUEL		56,776	5.3
50 CATAPULTING		---	---
51 MIN. FLYING WEIGHT		47,798	6.3
52 LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)		8	

53 WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)

54 STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)

55 PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.) 0

56

57 AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.)

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ at ℄ airplane.
***Parallel to ℄ airplane.
****Total usable capacity.

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TABLE V - 11

STRUCTUAL AND DESIGN DATA

USED FOR WEIGHT ESTIMATION

GEOMETRIC DATA WING

	Weight Prediction Value
Length of MAC, Inches	153
Area - Gross - Ft.^{2}	838
Area - Exposed - Ft.^{2}	655
Span - Gross - Ft.	65.75
Span - Exposed (One Side) Ft.	25.8
Span - Structural - Exposed (One Side) Ft.	25.8
Aspect Ratio	5.16
Taper Raio	1.0
Root Chord - Aircraft - Ft.	12.75
Root Chord - Exposed Area - Ft.	12.75
Tip Chord - Ft.	12.75
Root Chord Thickness Ratio	.21
Tip Chord Thickness Ratio	.21
Root Chord Thickness, Gross Area - Ft.	2.68
Root Chord Thickness, Exposed Area - Ft.	2.68
Tip Chord Thickness - Ft.	2.69
Torque Box Area, Gross, Ft.^{2}	314
Torque Box Area, Exposed, Ft.^{2}	264

GEOMETRIC DATA WING (Continued)

	Weight Prediction Value
Leading Edge Area, Exposed, Ft.^{2}	99
Trailing Edge Area, Exposed, Ft.^{2}	296
Ailerons Area Ft.^{2}	58
Trailing Edge Flaps (Type: Area, Ft.^{2})	108
Leading Edge Sweep Angle - Degrees	0
25% Chord Sweep Angle - Degrees	0

HORIZONTAL TAIL

Length of MAC - Inches	98
Area, Gross - Ft.^{2}	257
Area, Exposed, Ft.^{2}	233
Span, Gross, Ft.	32
Span, Exposed (One Side) Ft.	15
Span, Structural, Gross, Ft.	32
Span, Structural, Exposed, (One Side) Ft.	15
Aspect Ratio	4.0
Taper Ratio	0.7
Root Chord, Gross Area, Ft.	9.5
Root Chord, Exposed Area, Ft.	9.15
Tip Chord, Ft.	6.66
Root Chord Thickness Ratio	.15
Tip Chord Thickness Ratio	.15
Root Chord Thickness, Gross, Area, Ft.	1.43
Root Chord Thickness, Exposed, Area, Ft.	1.37

HORIZONTAL TAIL (Continued)

	Weight Prediction Value
Tip Chord Thickness, Ft.	1.0
Elevator (Movable Surface) Area, $Ft.^2$	68.7
Tail Moment Arm, 25% Wing MAC to 25% Horizontal Tail MAC - Ft.	38.83

VERTICAL TAIL

Number of Surfaces	1
Length of MAC Inches	153
Area, Gross, $Ft.^2$	141
Area, Exposed, $Ft.^2$	141
Span, Gross, Ft.	11.42
Span, Exposed, Ft.	11.42
Span, Structural, Gross, Ft.	11.42
Span, Structural, Exposed, Ft.	11.42
Aspect Ratio	0.89
Taper Ratio	0.6
Root Chord, Gross Area, Ft.	16.08
Root Chord, Exposed Area, Ft.	16.08
Tip Chord, Ft.	9.65
Root Chord Thickness Ratio	.15
Tip Chord Thickness Ratio	.15

VERTICAL TAIL (Continued)

	Weight Prediction Value
Root Chord Thickness, Gross Area, Ft.	2.51
Root Chord Thickness, Exposed Area, Ft.	2.51
Tip Chord Thickness, Ft.	1.45
Tail Moment Arm, 25% Wing MAC, to 25% Vert. Tail MAC, Ft.	31.42
Location of Horizontal Tail, Distance From Root Chord, Ft.	11.5

FUSELAGE

Overall Length, Ft.	68.33
Overall Width, Ft.	9.7
Overall Height, Ft.	11.5
Basic Structure Length, Ft.	63.0
Basic Structure Width, Ft.	9.7
Basic Structure Height, Ft.	11.5
Wetted Area (Total - $Ft.^2$)	2700
Pressurized Volume (PSI Differential $Ft.^3$)	0

LANDING GEAR

Type	Tricycle
Main Gear	Tandem
Nose Gear	Dual
C.B.R.	4
Number of Main Gear Wheels	2/Side
Number of Nose Gear Wheels	2
Sink Speed, Ft./Sec.	12

PROPULSION

	Weight Prediction Value
Number of Engines	4
Engine Type	Turbo-Shaft
Power Per Engine	5297
Nacelle Type	Tilting
Fuel System	
Tanks	
Number/Location	12/Wing
Capacity - Gals.	1,573
Type/Material	S/S .50 Cal.
Type Fuel/Density	6.5#/Gal.
Lubricating System	
Tanks	
Number	0
Capacity - Gals.	0
Coolers	
Number	0
Drive System	
Design Horsepower	21186
Propeller/Rotor RPM	206/Cruise
Engine RPM	6920 Cruise

ROTORS

	Weight Prediction Value
Type	Hingeless
Design Horsepower - Cruise/Hover	7580/5860
Tip Speed Ft./Sec.- Cruise/Hover	595/850
Blade Radius Ft.	27.5
Blade Chord Ft.	2.65
Number of Blades	3
Blade Area $Ft.^2$	72.82
Solidity	.092
Point of Blade Attachment, Distance from Centerline of Hub to Blade Attachment, Ft.	2.0625

TABLE V-12

1972 TECHNOLOGY MATERIAL STUDIES WEIGHT SAVING SUMMARIES WEIGHT REDUCTION IN PERCENT

Functional Group	SRR 4 [1]	ASD-TR-696 [2]	Seattle [3]	Wgt. Pred. Workshop [4]	Remarks
Wing Group	N/A	32.8%	25.0%	20-42%	
Box		37.1%			
Sec. Struc.		25.7%			
Control Surf.		26.3%			
Tail Group	N/A	16.0%			
Horizontal		21.2%			
Vertical		-----			
Body Group	11.5%	7.6%	21.5%		
Basic		10.3%			
Secondary		6.4%			Includes 11.8% in Nacelle and Boom Structure
Alighting Gear	1.0%	26.7%			

1. "Boron Program - Applications Analysis of the CH-46 and CH-47", SRR4, R. White, 5/3/67, Boeing-Vertol. (Reference V-1)

2. "Determination of Increased Aircraft Performance by Application of Composite Materials", ASD-TR-69-6, Vol. II, D.N. Ulry, October 1968. (Reference V-2)

3. "Weight Savings with Advanced Filament Composite Materials", Rough Draft of Boeing-Seattle Document, R.D. Martin, 10-5-66. (Reference V-3)

4. "Proceedings of the Fourth Weight Prediction Workshop for Advanced Aerospace Design Projects", Paper VI, "Advanced Composite Wing Structures", W. Ludwig, October 1968. (Reference V-4)

7. WEIGHT ESTIMATION SUBSTANTIATION

The detailed methodology used to derive the component weights for Model 215 are presented in this section.

As previously mentioned, the weights were determined through VASCOMP. Further verification of these weights will be accomplished during in-depth system design studies in Phases II and IV of this contract.

The weights are based on a 1972 state-of-the-art and reflect the consideration of advanced materials and advanced drive system technology.

It has been assumed that the overall weight of the wing, tail and body can be reduced by 12.5% and the nacelle can be reduced by 9.0% from 1969 Technology.

An "in-house" survey of previous advanced material studies has been conducted and the results of weight savings have been summarized in Table V-12.

A. <u>Prop/Rotors</u>

Weights as derived by VASCOMP are based on the emperical equation shown below. The constant, 13.5, reflects the utilization of a titanium hub and fibre-glass blades.

$$W_R = 13.5\ (K)^{0.67}$$

$$K = (r)^{0.25}\left[\frac{HP_R}{100}\right]^{0.5}\left[\frac{V_{TL}}{100}\right]\left[\frac{GA}{10}\right]$$

Where:

W_R	=	Weight of One Rotor	
r	=	Center Line of Rotation to Blade Attachement	Ft. = 2.063
HP_R	=	Horsepower/ Rotor	= 10593x1.1
V_{TL}	=	Design Tip Speed	Ft/Sec.= 850x1.1
σ	=	Solidity	= 0.092
R	=	Radius	Ft. = 27.5
A	=	Disc Area	$Ft.^2$ = 2375

Total Rotor Weight =

Rotors 2627.5 x 2 =	5,255
Spinners =	200
	5455#

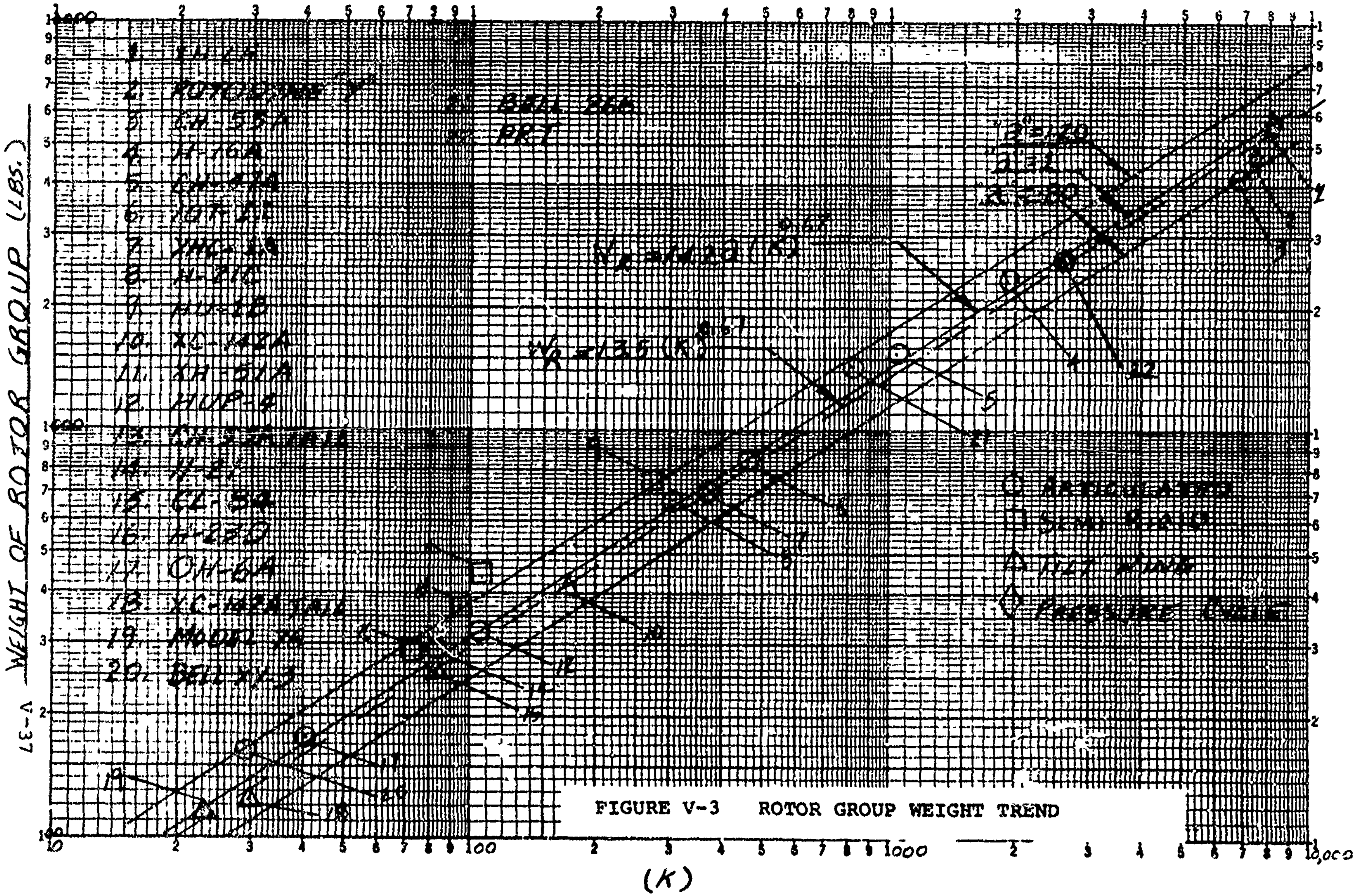


FIGURE V-3 ROTOR GROUP WEIGHT TREND

B. Wing

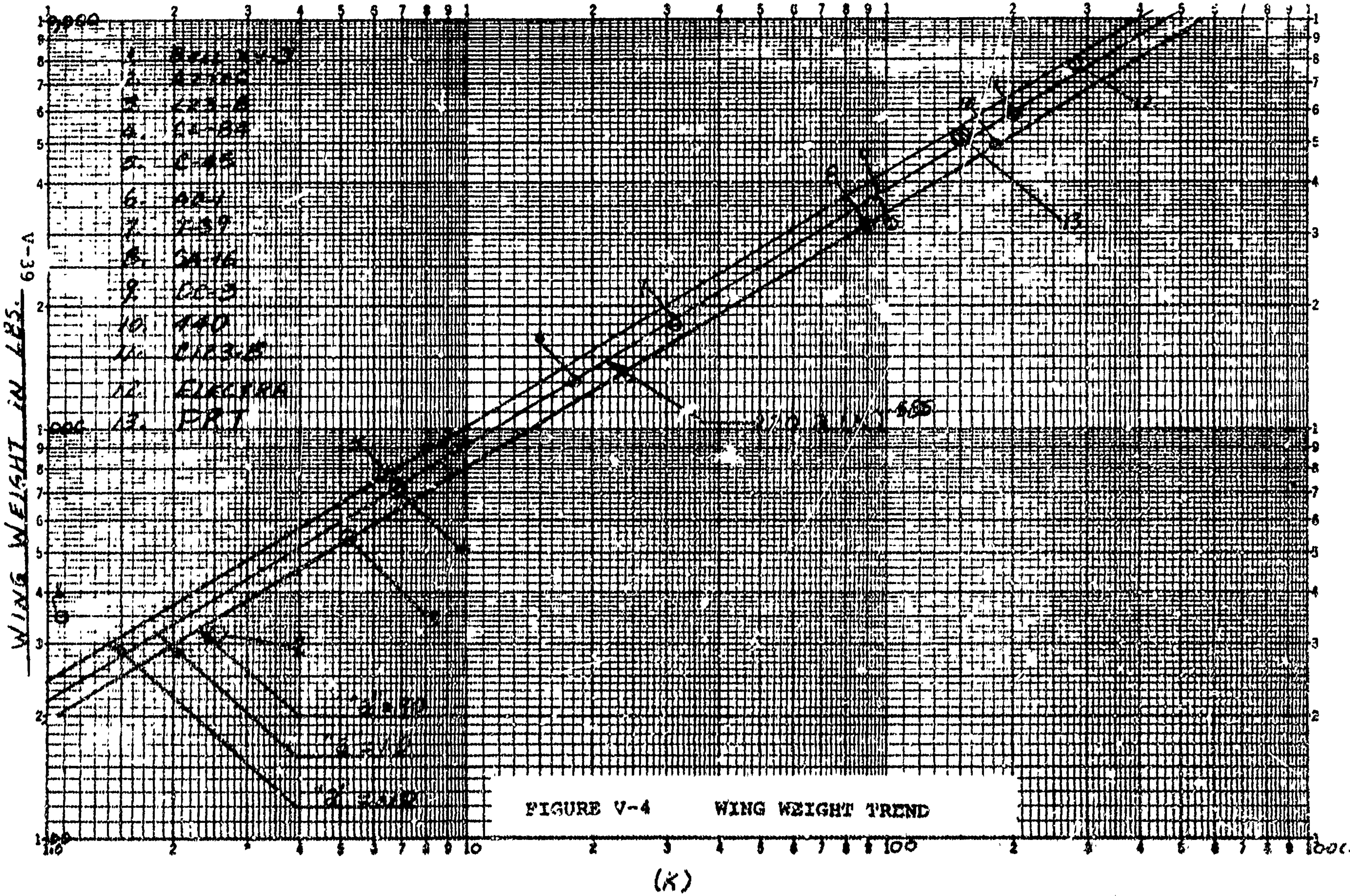
The emperical equation shown below is the basis for the weight as derived by VASCOMP. It has been assumed that, since the wing will be designed by the vertical flight conditions rather than forward flight conditions, the constant of 220 is increased by 25% to 275.

$$W_W = C\ (K)^{0.585}$$

$$K = \left[\frac{R_M\ W_X}{10^4}\right] \left[\frac{S_W}{10^2}\right] \left[\text{Log}\ \frac{b}{B}\right] \left[\frac{1}{2} + \frac{}{K_r}\right] \left[\sqrt{N}\right] \left[\text{Log}\ V_D\right] \left[\text{Log A.R.}\right]$$

Where:

W_W	- Weight of Wing		
R_m	- Relief Term		= 0.89
W_x	- Body Contents Weight	Lb.	= 26,576
S_W	- Gross Platform Area of Wing	$Ft.^2$	= 838
b	- Wing Span		= 66.4
B	- Max Fuselage Width		= 9.7
	- Taper Ratio		= 1.0
K_R	- t/c at Wing Root		= 0.21
N	- Ultimate Load Factor		= 4.5
V_D	- Dive Velocity	kts.	= 414
AR	- Aspect Ratio		= 5.26


WING WEIGHT IN LBS.
(K)
8. SA-16
9. DC-3
10. 440
11. C123-B
12. ELECTRA
13. PRT
FIGURE V-4 WING WEIGHT TREND

W_w = 5077

1972 Material Factor (Less 12.5%) -632

Add Pod Attachment Fittings +500

Total Wing Weight 4,945#

C. Horizontal Tail

The weight of the horizontal tail was derived from the following equation. Since a unit tail is used, the weight can be further reduced. This weight saving has not been incorporated here.

$$W_{HT} = 360\,(K)^{0.54}$$

$$K = \left[F_H\right]\left[\frac{S_H}{10^2}\right]\left[\frac{\text{Log } V_D}{TMA \times t}\right]$$

$$F_H = \left[\frac{W_G}{10^4}\right]\left[\frac{K_y}{10}\right]\left[\frac{b_H}{10}\right]\left[\frac{1 + 2\lambda_H}{1 + \lambda_H}\right]$$

W_{HT}	= Weight of Horizontal Tail		
S_F	= Planform Area	$Ft.^2$	= 258.6
V_D	= Dive Speed	KTS.	= 414
TMA	= Tail Moment Arm	Ft.	= 38.8
t	= Root Thickness	Ft.	= 1.4
WG	= Design Gross Weight	Lbs.	= 67,000
Ky	= Pitch Radius of Gyration	Ft.	= 10.8
b_H	= Span	Ft.	= 32.2
λ_H	= Taper Ratio		= 0.7

Horizontal Tail Weight = 762#

1972 Material

Factor (Less 12.5%) = -95

TOTAL HORIZONTAL TAIL = 667#

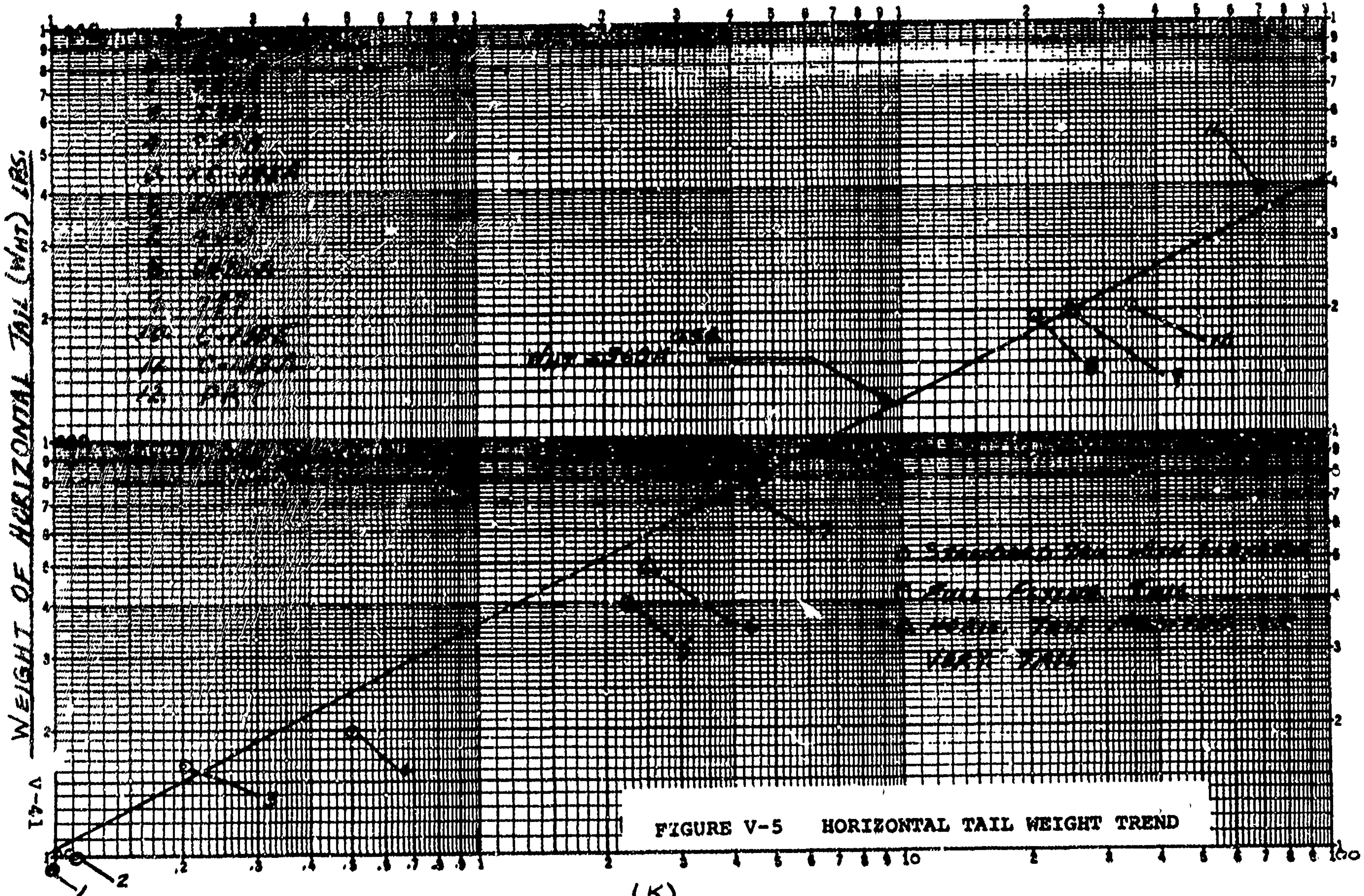


FIGURE V-5 HORIZONTAL TAIL WEIGHT TREND

D. Vertical Tail

The VASCOMP weights were derived from the following equation:

$$W_{VT} = 380\ (K)^{0.54}$$

$$K = \left[F_V + \frac{a\ F_H}{2\ b_V}\right]\left[\frac{S_V}{10^2}\right]\left[\frac{\text{Log}\ V_D}{\text{TMA} \times t}\right]$$

$$F_V = \left[\frac{W_g}{10^4}\right]\left[\frac{K_Z}{10}\right]\frac{b_V}{10}\left[\frac{1+2\ V}{1+\ V}\right]$$

Where:

W_{VT} = Weight of Vertical Tail

a = Distance From Horizontal Tail to Root of Vertical Tail — Ft=13.0

b_V = Span — Ft=13.0

S_V = Area — Ft^2=141

V_D = Dive Speed — kts=414

TMA = Tail Moment Arm — Ft =31.4

t = Tickness at Root — Ft =2.04

W_g = Design Gross Weight — lbs=67,000

K_Z = Yaw Radius of Gyration — Ft=23.3

b_V = Span — Ft=13.

= Taper Ratio — = 0.6

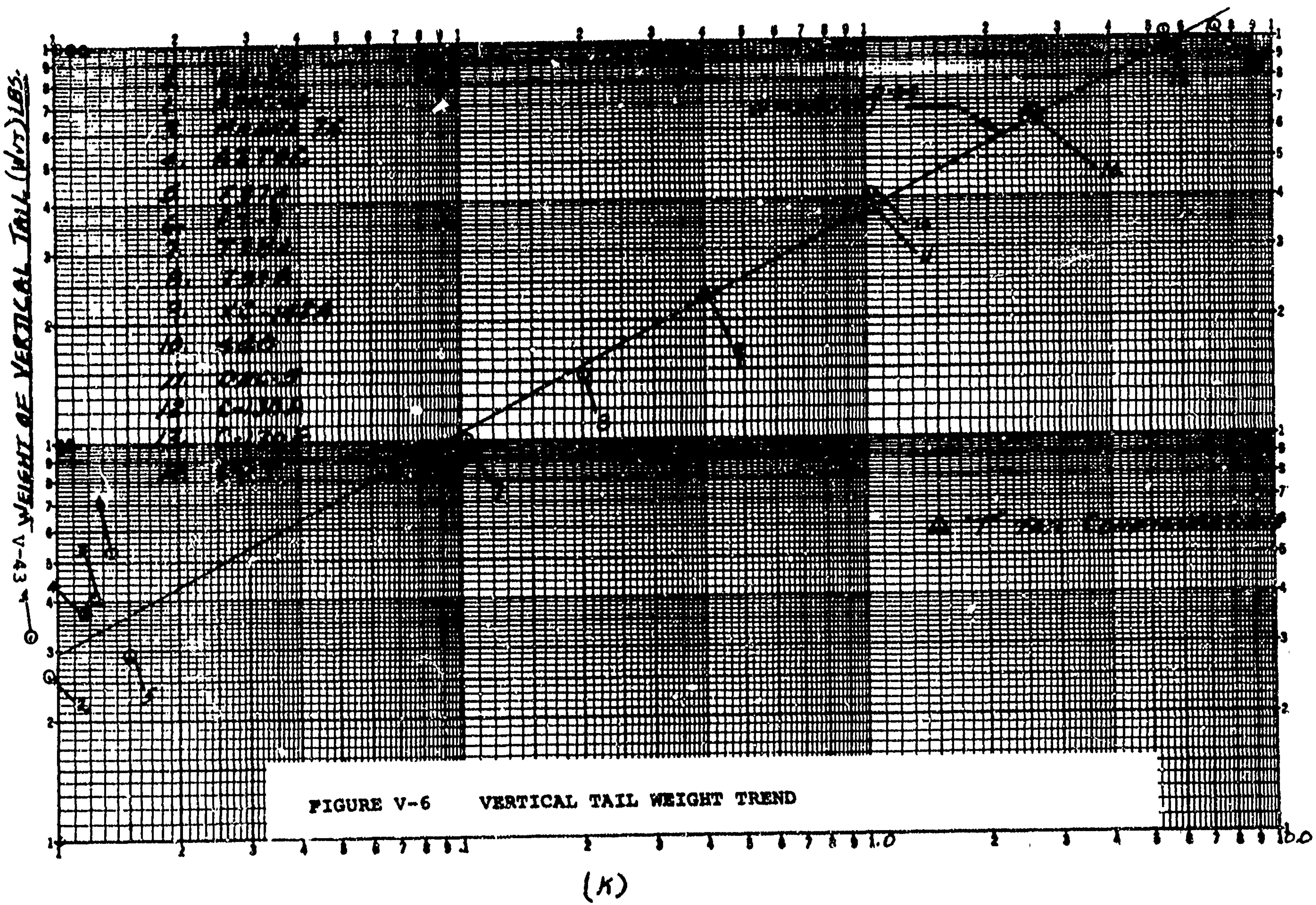


FIGURE V-6 VERTICAL TAIL WEIGHT TREND

Vertical Tail Weight - 623

1972 Material
Factor (Less 12.5%) - -77

TOTAL VERTICAL TAIL - 546 Lbs.

E. Body

Weights as derived by VASCOMP were obtained through the equation shown below. It will be noted on the trend curve, Figure V-7, that the weight is slightly higher than those of other transport aircraft. Previous studies on bodies in this class indicate this trend.

$$W_B = C \quad (K)^{0.508}$$

$$K = \left[\frac{W_X}{10^4}\right]^{0.7} \left[\frac{S_f}{10^3}\right] B \left[L_f + L_{RW}\right]^{0.5} \left[\text{Log } V_D\right]^{0.2} P+1 \quad N^{0.3}$$

Where:

W_B	=	Weight of Body	
C	=	Constant	128
W_X	=	Body and Contents Weight	lb=26,576
S_f	=	Wetted Area of Body	Ft^2=2,280
B	=	Maximum Body Width	Ft=9.7
L_f	=	Body Length - Basic	Ft=63.0

L_{RW}	=	Length of Ramp Well	Ft =	20.7
V_D	=	Dive Speed	kts=	414
P	=	Limit Differential Cabin Pressure	psi=	0
N	=	Ultimate Load Factor		4.5

W_B = Body Weight - 6280

1972 Material Factor (-12.5%)	- 783
TOTAL BODY WEIGHT	5497 Lbs.

ADD: 463 L Cargo Loading System 980

CABIN	(860)
Side Rails	94
Roller Trays	151
Rollers & Shafts	78
Pallet Locks	140
Master Lock Control	8
Winch	300
Crash Net	89

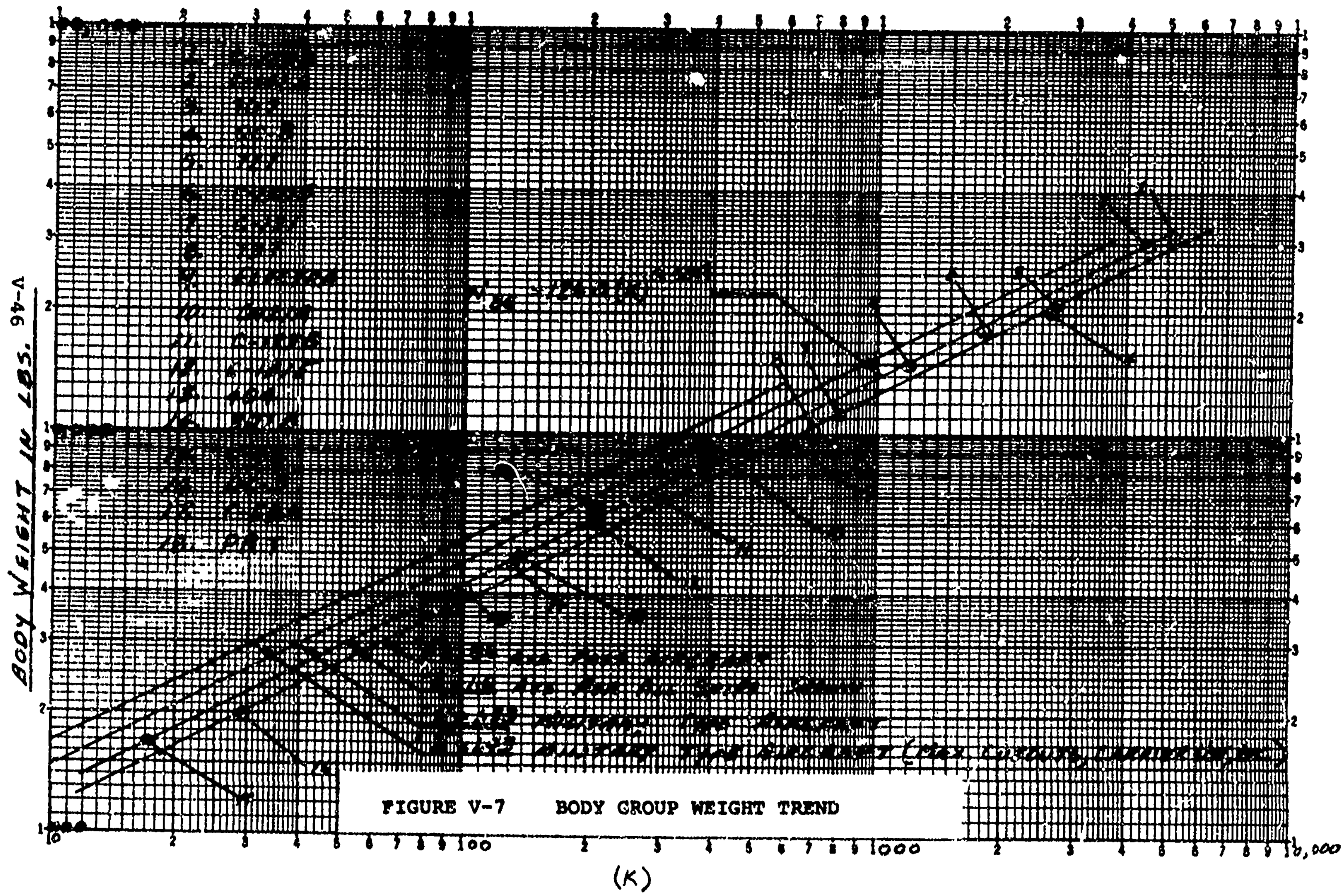


FIGURE V-7 BODY GROUP WEIGHT TREND

RAMP	(120)
Side Rails	34
Roller Trays	55
Rollers and Shafts	29
Teeter Rollers	3
TOTAL BODY	6,477 Lbs.

F. Landing Gear

The weight of the landing gear has been based on 3.8% of Design Gross Weight. This includes wheels, brakes, tire tube, struts, linkages, retracting mechanism and pods. No penalty has been assigned for rough field STOL take-off. Therefore, the weight reflected here is for STOL take-offs from semi-prepared (e.g. landing mats) and paved runways.

The basic design criteria are a sink speed of 12 feet per second, and CBR = 4.

Table V-13 is a tabulation of V/STOL landing gear weights in per cent of gross weight and it shows that STOL aircraft typically have a higher weight landing gear than primarily VTOL aircraft. Further reduction in landing gear weight can be realized through use of better high strength materials. These reductions have not been incorporated at the time.

TABLE V-13

SUMMARY OF

LANDING GEAR WEIGHT
IN
PERCENT OF GROSS WEIGHT

FOR

V/STOL AIRCRAFT

Helicopters	% G.W.	Airplane	% G.W.
CH-46A	3.1	Bell XV-3	3.1
CH-46D	2.8	XC142A	3.2
CH-46E	3.1	Bell 266	3.6
CH-47	3.4		
CH-47C	3.3	*DeHavalland	
CH-3C	3.4	DHC-5	4.2
CH-53A	2.9	*Brequet.941S	4.5
CH-54	4.7	*DeHavalland	
CH-54A	4.7	DHC	5.4
107-2	3.1	*C130	4.1
AH-56A	3.6	*C123	4.3
HH-52A	5.9		
HUP-2	3.2		
UH-34D	3.7		
SH-3A	4.2		
H-21C	3.6		

*Rough Field Requirements

G. Flight Controls

Flight controls include all controls for Propeller/Rotor, cockpit, conventional airplane controls, tilt mechanism and the stability augmentation system. The values used for K are a result of previous studies.

Cockpit Controls	
$W_{cc} = K_{cc} \left[\frac{WG}{1000}\right]^{0.41}$	
$K_{cc} = 26.$	145
Rotor Upper Controls	
$W_{UC} = K_{UC} \; W_R$	
$K_{UC} = 0.45$, W_R = Weight of Rotors	2,367
Rotor Hydraulics	
$W_H = K_H \left[\frac{WR}{100}\right]^{0.84}$	
$K_H = 30.$	836
Conventional Airplane Controls	
$W_{CA} = K_{CA} \left[W_G\right]$	
$K_{CA} = 0.013$	871
Tilt Mechanism	
$W_{TM} = K_{TM} \left[W_G\right]$	1005
$K_{TM} = 0.015$	
Stability Augmentation System	175
TOTAL FLIGHT CONTROLS	5,399

Drive System

The weight of the drive system includes gear boxes, cross shafting, lubrication, etc. The constant in the equation, 220, reflects a 15% reduction due to advanced drive system technology wherein a bending stress of 40,000 psi, and a Hertz stress of 180,000 psi for spur helical gear teeth and 260,000 psi for spiral and bevel gear teeth is anticipated using VASCO x2 modified vacuum melt alloy steel.

Present designs use SAE 9310 carbonized steel with a bending stress of 30,000-34,000 psi and a Hertz stress of 150,000-160,000 psi for spur and helical gears and 225,000-250,000 psi for spiral and bevel gears.

The equation used by VASCOMP is

$$W_{DS} = C \left[K \right]^{0.8}$$

$$K = \left[\frac{H.P._{Total}}{RPM\ Rotor} \right]^{1.1}$$

Where:

C	=	Constant	= 220
HP_{Total}	=	Total Horsepower	= 21,186
RPM_{Rotor}	=	Rotor Design RPM	= 295

Total Drive System Weight = 7,209 Lbs.

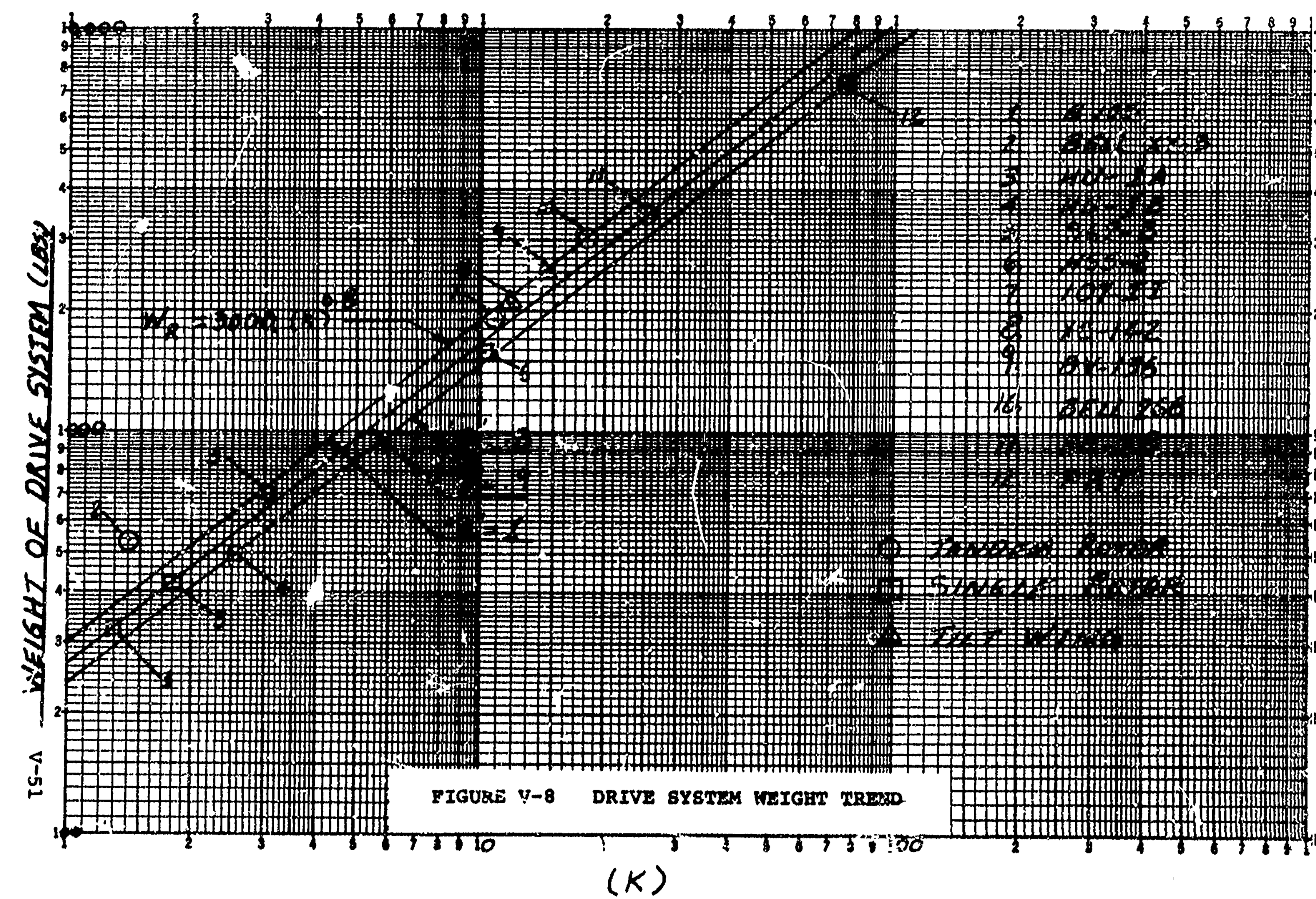


FIGURE V-8 DRIVE SYSTEM WEIGHT TREND

I. Engines

The engines considered are discussed in Section III. The weights are based on .116 pounds per horsepower.

J. Engine Installation

VASCOMP determines the weight of the engine installation in terms of percent of engine weight. Engine installation includes air induction, exhaust, cooling, lubricating, water injuection, and engine controls.

Previou studies on engine installation of this type indicate this factor to average 40%.

The estimated breakdown of this is:

Air Induction	308
Exhaust	390
Lubricating	30
Controls	90
Starting	195

K. Fuel System

The fuel system for this aircraft consists of crash-restraint self-sealing tanks (protected from .50 caliber gun fire), pumps, plumbing, etc. The basis for the weight in this study is on a pound per gallon value.

W_{FS} = 1#/Gal x 1573 = 1,573

Inflight Re-fueling = 63

TOTAL FUEL SYSTEM = 1,636

L. Nacelle Structure and Fairing

VASCOMP determines the weight of the nacelle on a per cent of engine weight basis. Previous studies on this type of installation indicate the nacelle weight to be 65% of the engine weight.

W_n = .65 (2,543) = 1,653

1972 Material Factor (9.0%) = -148

TOTAL NACELLE WEIGHT 1,505#

M. Fixed Equipment

The fixed equipment weights are listed in Table VI and are based on estimated systems. The hydraulic and electrical groups were based on a percentage of gross weight, as shown, and broken into sub-groups as an estimate only.

TABLE V-14

TABULATION OF FIXED EQUIPMENT WEIGHTS FOR MODEL 215

Auxiliary Power Plant	200.0
Engine	91.0
Eng. Supports	5.0
Air Induction	5.0
Exhaust System	5.2
Lube System	2.0
Fuel System	8.0
Controls	8.7
Starting System	56.4
Insul. and Blankets	18.7

Instruments & Navigation				300.0
Flight	Ind	XMTR	Instl.	46.4
Altimeter (2)	3.6			3.6
Airspeed (2)	2.0			2.0
Vert Speed (2)	3.2			3.2
Height Ind. (2)	4.6			4.6

Instruments & Navigation (Continued)

Flight	Ind	XMTR	Instl	46.4
Compass - Mag.	.7		.6	1.3
Free Air Temp.	.8	.3	.1	1.2
Eng. & Flap Pos. (2)	2.2	.1	.9	3.2
Rudd. & Ailer Net	.4	1.6		2.0
Land. Gr. Pos.	1.0	.9	2.0	3.9
Clock-Mech.	1.1			1.1
Stall Warning (2)	3.6	3.0	3.8	10.4
Pitot Static			9.9	9.9
Propulsion				**221.4**
Fuel - Quantity	1.3	30.0	53.8	85.1
Flow	4.8	4.5	20.1	29.4
Engine-Turbine RPM	3.1	2.8	5.1	11.0
Inlet-Temp. (2)	2.3		16.0	18.3
Torque (2)	3.2		7.5	10.7
Total Torque (2)	2.3		2.3	4.6
Engine Oil-Press	1.8	4.0	10.4	16.2
Temp (2)	1.5	.8	2.6	4.9
XMSN Oil Press	1.8	7.6	10.0	19.4
Temp	1.8	2.8	10.0	14.6
Level	.3	1.0	3.4	4.7
Propeller RPM	.8	.7	1.0	2.5
Miscellaneous				**32.2**
Hydraulic Press. (3)	.7	6.1	3.3	10.0
Master Caution	.5		5.0	5.5
Caution Panel	3.6		2.3	5.9
Ice Detector			6.2	6.2
De-Ice & Anti-Ice			3.0	3.0
Nose Trim (2)	.4			.4
Oxygen Quantity	.9		.3	1.2

HYDRAULICS 0.5% GW = .005 (67,000) = 335.0

Estimated Weights

Pump Motor	10.0
Reservoirs	12.0
Filters	7.0
Press. Reg.	2.0
Transfer Valve	16.0
Shut-Off Valve	2.0
Emergency Valve	9.0
WG. Trans.	1.0
Grnd. Test Ftg.	3.0
Controls	13.0
Plumbing	165.0
Fluid	42.0
Supports	53.0

ELECTRICAL 1.9% GW =0.019 (67,000) 1248

Estimated Weights

A.C. System	784
Gen (4)	180
C.S.D. Units (4)	120
Transformer (2)	24
Super Panels (3)	7
Pwr. Monitor	3
Main C/B Panels (5)	15
C/B Panels	1
Wiring & Plugs & Misc.	384
Supports	50

D. C. System	464
Battery	50
Battery Chg.	12
D.C. C.B. Pnls & Diodes (2)	1
Battery Relay	1
Sw. & J-Box	18
Wiring & Plugs & Misc.	292
Lights & Signals	60
Supports	30
ELECTRONICS	1093.0
Communications	105.2
HF-SSB-VHF-FM	42.5
UHF-AM	10.0
VHF-AM	10.0
Inter Com	14.5
P.A.	10.0
IFF (Aims)	18.2

<u>Navigation & Radar</u>	<u>421.6</u>
D.I.L.S.	100.0
Tacan	22.0
Radar Alt.	15.0
UHF/ADF	11.6
ILS with VOR	6.1
Station Keep	50.0
LF-MF/ADF	11.6
Multi-Mode Rad	188.0
Back-Up Head Ref.	17.3
<u>Computing</u>	<u>236.4</u>
Air Data Comp.	3.4
Aerial. Del.	100.0
Aids	133.0
<u>Crash Recorder</u>	<u>28.0</u>
<u>Avionics Instl.</u>	<u>301.8</u>
Antennas	65.8
Radomes	35.0
Wiring & Plugs	164.6
Supports	36.4

ARMAMENT		**50.0**
Provisions for Armor Plate		50.0
FURNISHINGS & EQUIPMENT		**1812**
Personnel Accom.		699.0
Pilot & Co-Pilot		70.0
Seats (2)	23.0	
Seat Belts (2)	6.0	
Harness & Reel (2)	11.0	
Adjust Mech.	6.0	
Tracks & Supts.	24.0	
Crew Chief		19.0
Seat	11.0	
Seat Belt	3.0	
Harness & Reel	3.0	
Tracks & Supt.	2.0	
Troops (60)		434.0
Seats	200.0	
Belts	64.0	
Tracks & Supts.	170.0	
Misc. Pers. Accom.		75.0
Litter Instl.	71.0	
Relief Tube	4.0	
Oxygen System		101.0
Lox Conv.	25.0	
Fixed Prov.	71.0	
Prov. Recharging	5.0	

FURNISHINGS & EQUIPMENT (Continued)

Misc. Equipment		125.0
Windshield Wiper & Washer		29.0
Instr. Boards		17.0
Consoles		18.0
Overhead Consoles		10.0
Tie-Down Fittings		51.0
Furnishings		865.0
Soundproofing		865.0
Cockpit	50.0	
Cabin Forward	125.0	
Cabin, Propeller Plane	110.0	
Cabin Aft	465.0	
Ramp	115.0	
Emergency Equipment		123.0
Fire Extg.		55.0
Controls		9.0
Plumbing		14.0
Wiring & Instl.		10.0
Eng. Fire Det.		7.0
APU Fire Det.		1.0
Cockpit Fire Extg.		8.0
Cabin Fire Extg.		16.0
First Aid Kit		3.0

AIR CONDITIONING & DE-ICING		**394.0**
Air Conditioning		**255.0**
A/C Unit		103.0
Ducting		78.0
Plumbing		34.0
Controls		14.0
Supports		26.0
De-Icing		**139.0**
Wing & Tail		57.0
Airlines & Hoses	39.0	
Distribution	3.0	
Controls	8.0	
Electrical	4.0	
Supports	3.0	
Air Induction		58.0
Ducting	20.0	
Plumbing	18.0	
Controls	8.0	
Electrical	6.0	
Supports	6.0	
Prop & Spinner		18.0
Controls	8.0	
Electrical	10.0	
Canopy & Windshield		6.0

<u>AUXILIARY GEAR</u>	24.0
<u>A/C Handling</u>	24.0
Tie-Downs	5.0
Jacking	8.0
Towing	3.0
Hoisting	8.0

SECTION VI

FLYING QUALITIES

1. SUMMARY

A preliminary design evaluation of the transport configuration tilt rotor aircraft has been made which shows the need for stability augmentation. There appears to be no unusual inherently difficult flying qualities problems. It is suggested that most of the control and the stability augmentation for this aircraft be provided by rotor controls so that rotor blade load alleviation can be included. Vibration level of the present design is estimated to reach a maximum of 0.11 g at the helicopter end of transition.

2. CRITERIA

Flying qualities criteria to be applied to USAF Tilt Rotor aircraft design will be MIL-F-008785A (USAF) for flight at speeds above V_{CON} and the USAF-Cornell Aeronautical Laboratory proposed V/STOL flying qualities criteria, Reference VI-1, with speeds up to and including V_{CON}. For this effort V_{CON} is defined as that airspeed at which a load factor of 1.2 can be achieved with the wing flaps retracted and with no lift produced by the rotors. It is assumed that all approaches to landings will be made in the transition flight mode with the V/STOL criteria applicable. The aircraft has been assumed to be of Class II (heavy utility/search and rescue or assault transport) and has been evaluated for Category B flight phases.

Vibration criteria of MIL-H8501A indicates that 0.15 g at the number of blades per rev frequency shall not be exceeded at speeds below cruise speed. The present design will comply with this criterion but a more stringent criterion is believed necessary. Ground handling and ground resonance stability will be as defined in Reference VI-1 or MIL-H8501A.

3.0 INTEGRATED LOAD ALLEVIATION AND FLIGHT CONTROL SYSTEM

Recent developments in flight control systems have shown the advantages of compromising the stability augmentation system to reduce structural fatigue loads. The Load Alleviation by Modal Suppression (LAMS) system developed by Boeing for the B-52H is the production example of such a system. This concept is of considerable value for the hingeless rotored tilt rotor aircraft since the first bending mode rotor blade stresses are easily suppressed in all flight modes by rotor cyclic pitch control. Such a Load Alleviation by Rotor Modal Suppression (LARMS) system is assumed to be used on the USAF Tilt Rotor Model 215 aircraft.

As presently conceived, the LARMS system will provide feedbacks to alleviate problems of gust sensitivity, all of the known rotor-airframe stabilities and airframe elastic effects on flying qualities as well as rotor blade stresses. Also, stability augmentation will be provided in pitch and yaw in the helicopter mode and to damp the dutch-roll airplane mode. This system mainly consists of bi-cyclic rotor controls with nacelle-moment feedback.

This system has five major advantages for the tilt rotor configuration which are:

a. Tail surfaces can be sized for minimum stability since the static destabilizing effects of the prop-rotors will be canceled by the system.

b. Design of the wing and nacelle structure does not have to be compromised for increased stiffness to avoid instabilities and/or flying qualities problems due to wing twisting caused by rotor moments.

c. Design of the landing gear to damp ground resonance oscillations will not be as critical.

d. A nacelle tilt synchronization system is not required but would be provided for fail-safety.

e. The elevator and rudder surfaces airplane controls can be eliminated by using the rotor controls. Ailerons must be retained.

Control logic schematic for pitch, roll and yaw attitude controls are given in Figure VI-1, VI-2 and VI-3, respectively. These controls will provide the following functions.

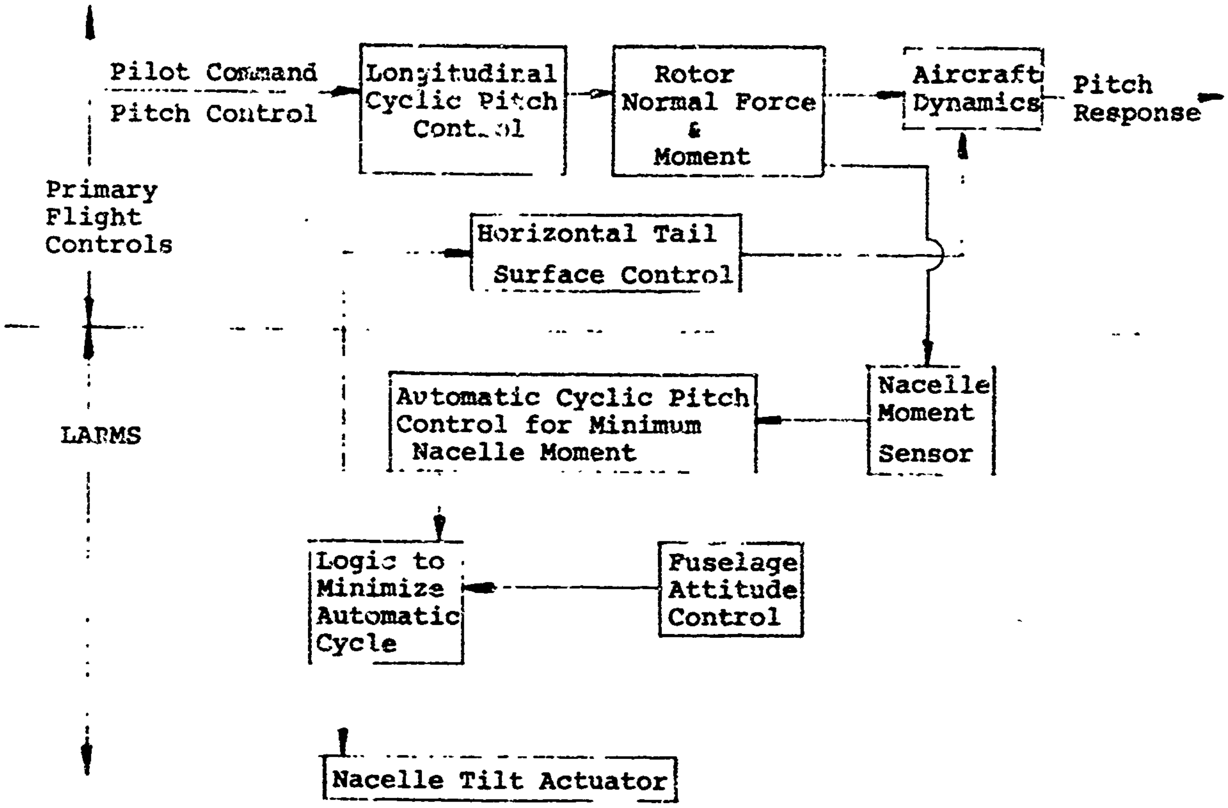
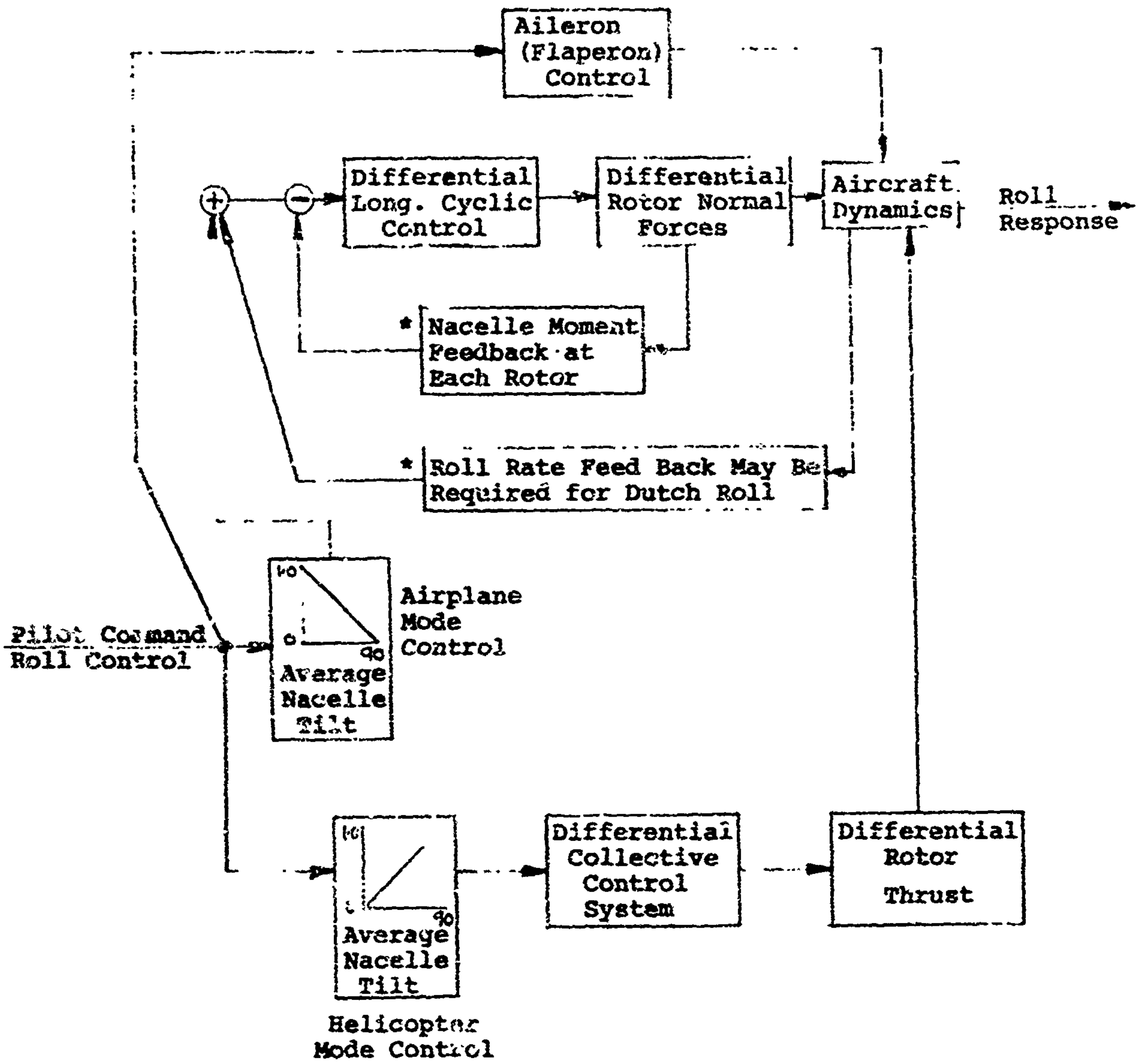


FIGURE VI-1 CONTROL LOGIC SCHEMATIC - PITCH ATTITUDE CONTROL



* These Feedbacks Drive
Parallel Automatic
(SAS) Actuators

FIGURE VI-2 CONTROL LOGIC SCHEMATIC - ROLL ATTITUDE

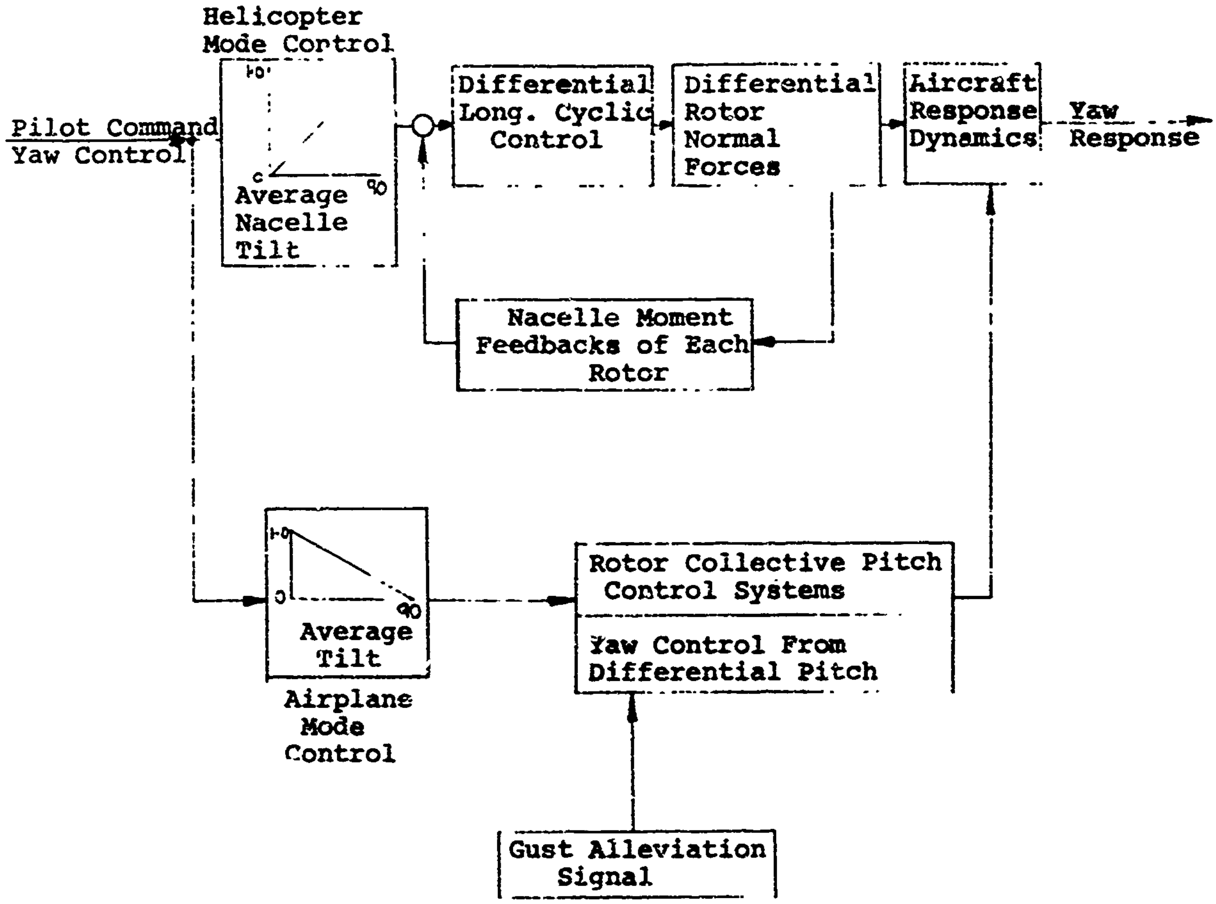


FIGURE VI-3 CONTROL LOGIC SCHEMATIC - YAW CONTROL

a. Bi-cyclic rotor controls and SAS actuators are provided for the LARMS controls. The lateral cyclics should have about the half of the authority of the longitudinal cyclics.

b. Fuselage attitude should be controlled in hover by an attitude sensor which commands the required automatic longitudinal cyclic pitch. This system needs to have response characteristics such that the pilot can cause aircraft pitch attitude changes for transition control.

c. Nacelle tilt should be driven by a nacelle moment feedback loop in such a way that the automatic longitudinal cyclic pitch is minimized. It is expected that this system will have a slow response.

d. Horizontal tail control should be limited to a slow response system driven by a feedback which acts to minimize the longitudinal rotor cyclic pitch in cruise.

e. Vertical and lateral gust sensitivity will be minimized by the cyclic controls with nacelle moment feedback which therefore must act in the cruise mode. An automatic collective pitch system will be required in cruise to prevent horizontal gust sensitivity. This feedback system requires a low sensitivity (small pitch change per unit acceleration) but a fast response.

Fuselage attitude control is provided by longitudinal cyclic pitch of the rotors with the aid of the horizontal tail trim in transition and cruise. In hover, moment trim is provided by the cyclic pitch and if the cg is at an extreme a few degrees of attitude change will be required to balance the longitudinal forces. When the nacelles tilt during transition and as the horizontal tail becomes effective, additional moments are produced which must be trimmed. An attitude sensor and control feedbacks could be provided as part of the control system to provide near zero trim attitude change with the use of a minimum cyclic pitch control. The pilot could be provided with a trim control to select the attitude he prefers but this is not expected to be a necessary pilot control function.

This LARMS control system, as shown in Figures VI-1, VI-2, and VI-3, could be provided with any of the advance flight control schemes at the pilot interface. This system could also be readily integrated with the avionics for navigation and position holding.

4. AIRCRAFT LIFT-DRAG POLARS

The lift and drag characteristics of the aircraft have been determined by using well-known and tested methodology and are shown in Figures VI-4. The effects of the geometric properties of the wing (high thickness/chord ratio, low aspect ratio and simple plain flap system) and the transport fuselage are evident in this figure. The aircraft lift polar, Figure VI-5 shows the $C_{L_{MAX}}$ produced by this flap system. The drag polar for this aircraft at low Mach number is given in Figure VI-6 and the effect of compressibility is shown in Figure VI-7. It is shown that the aircraft will have some drag divergence when flying at dash speed. This could be cured with further refinement of the design but probably would not be a problem unless some side effect such as aileron-buzz occurred.

The calculations for obtaining C_L and $C_{L_{MAX}}$ at varying angles of attack and flap settings, and consequently V_{STALL}, for the baseline configuration followed Section 4.1 of Reference VI-2. Starting from experimental low speed air-foil section aerodynamic characteristics, conventional corrections were applied to account for the effects of boundary layer influenced by wing surface form and roughness,

NUMBER
REV LTR

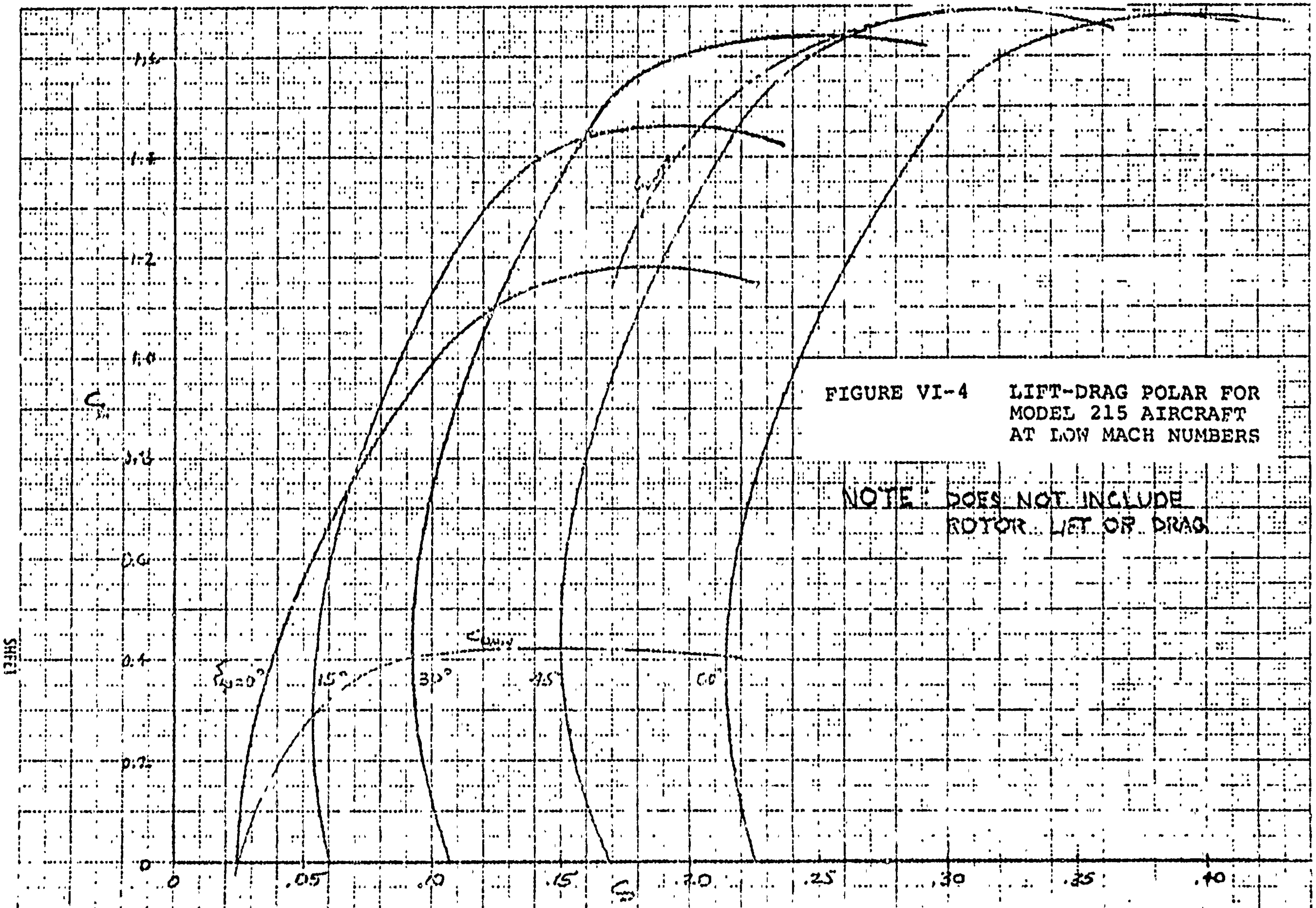


FIGURE VI-4 LIFT-DRAG POLAR FOR MODEL 215 AIRCRAFT AT LOW MACH NUMBERS

SHEET

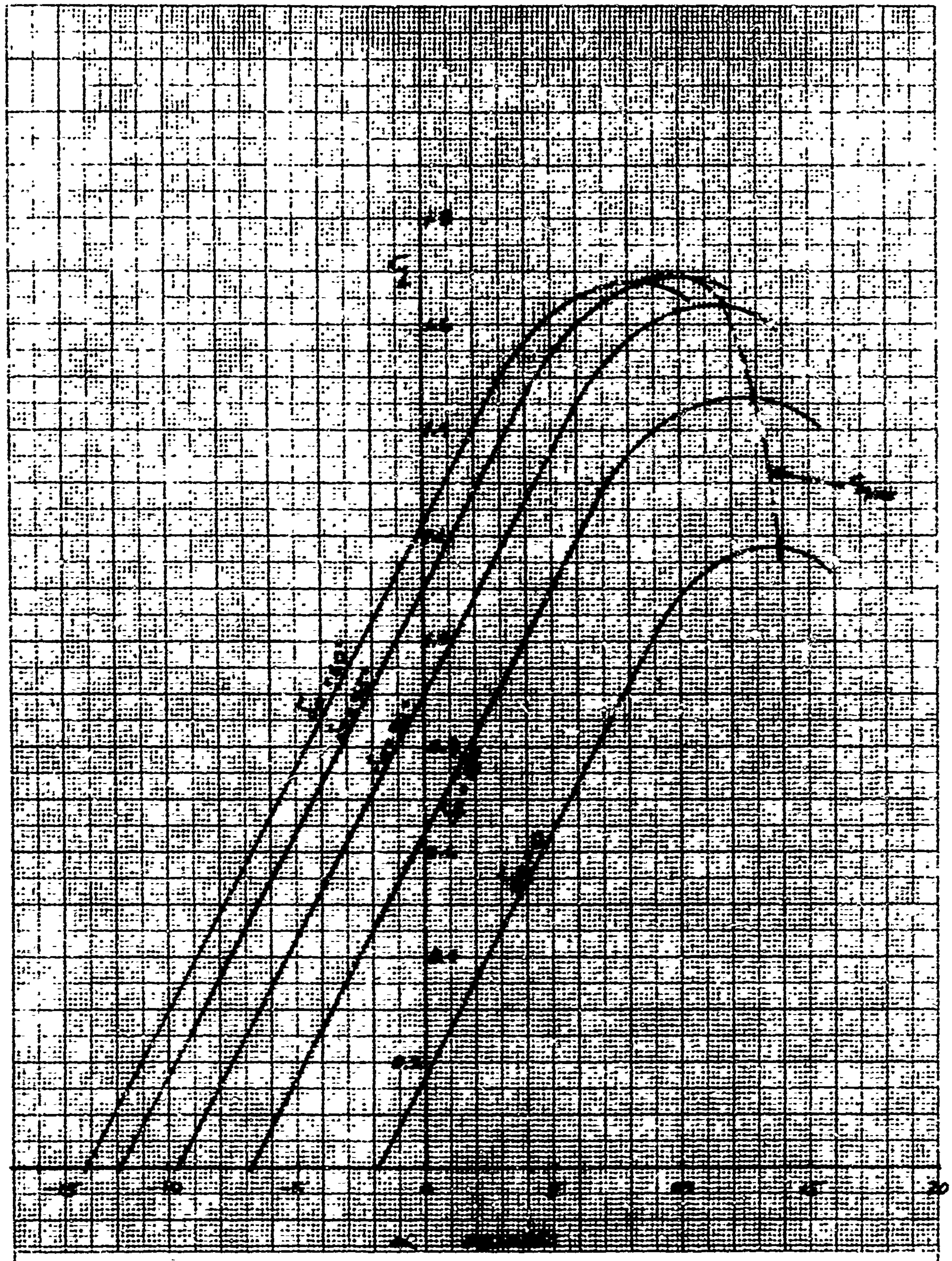


FIGURE VI-5 AIRCRAFT LIFT POLAR SHOWS GENTLE STALL CHARACTERISTICS

REV LTR

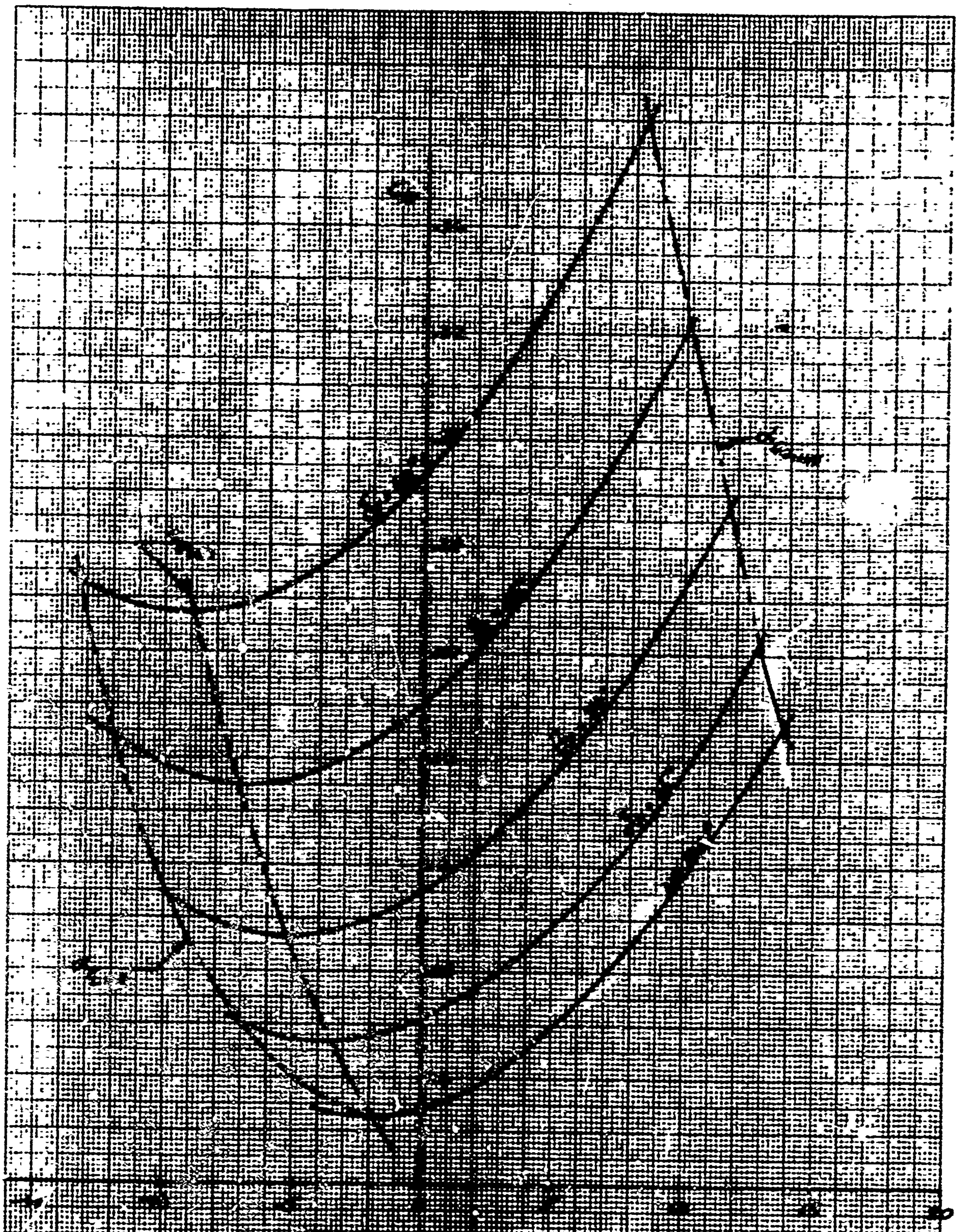


FIGURE VI-6   AIRCRAFT DRAG POLAR AT LOW MACH NUMBERS

SHEET

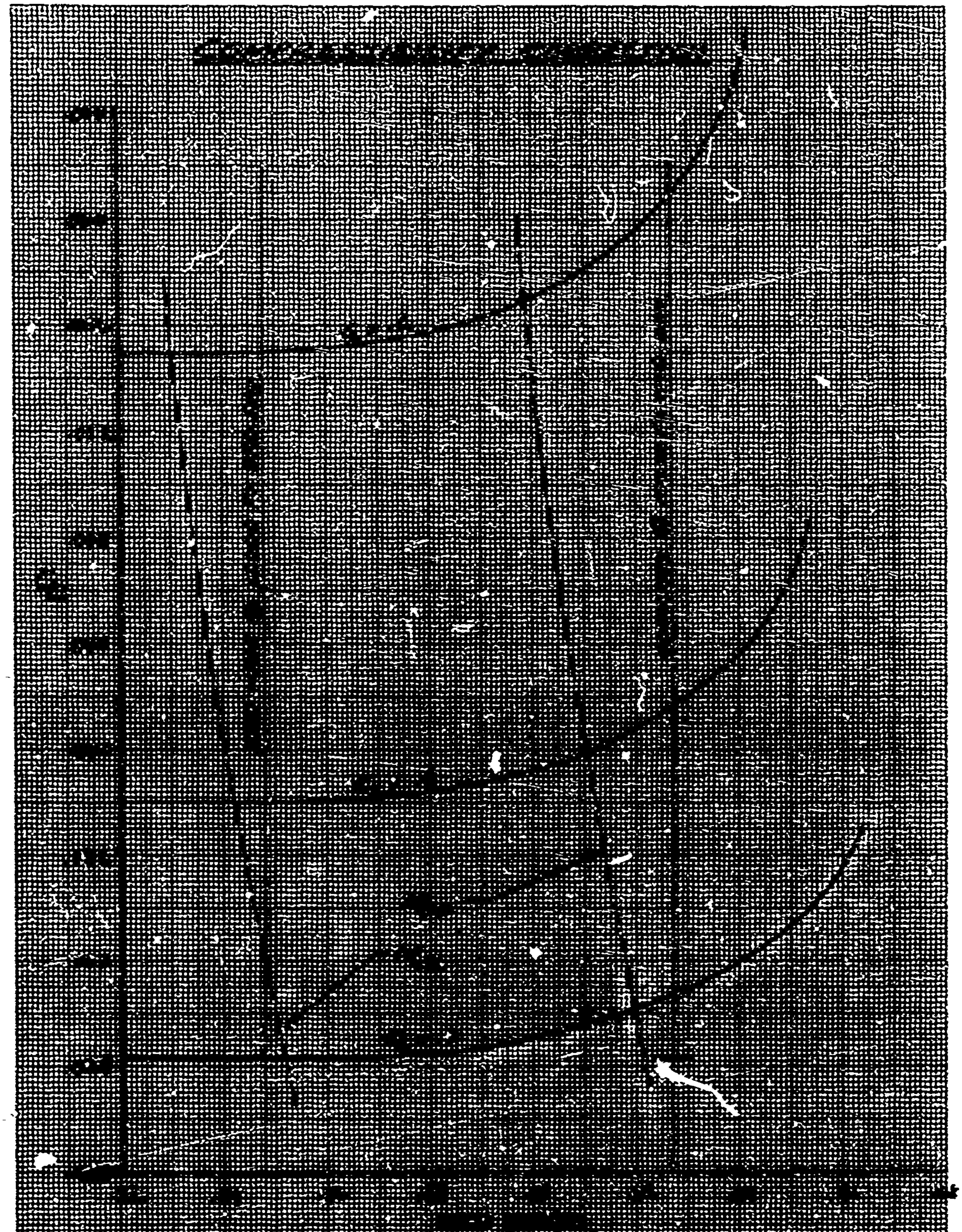


FIGURE VI-7 EFFECT OF COMPRESSIBILITY ON AIRCRAFT DRAG SHOWS DRAG DIVERGENCE AT MACH 0.62 WITH THICK WING AND WITHOUT ADVANCED AIRFOIL SECT.

as well as geometric arrangement and Reynolds Number. Reference VI-2 utilizes the method of Reference VI-3 which is especially applicable to the straight untapered, untwisted wing of the design-point aircraft for determining its three-dimensional lift-curve slope. This method gives results that agree with slender-wing theory at very low-aspect-ratios and with two-dimensional section data at infinite aspect ratios. The wing alone C_L obtained was then adjusted to reflect the presence of the fuselage. The wing-body contribution was obtained from wind tunnel test data of Reference VI-4. The contribution of the tail to the aircraft C_L was computed using the methods outlined in References VI-2 and VI-4. The parameters of the aircraft are given in Table VI-1.

For the thick airfoils utilized, stall usually occurs as a result of separation from the trailing edge and is characteristically mild, with gradual rounding of the lift and moment curves near $C_{L_{MAX}}$. The maximum lift of these sections is correlated by using the position of maximum thickness in addition to the y - parameter (the difference between the upper surface ordinates at the 6% chord and 1.5% chord stations, respectively). There is also a maximum lift increment

TABLE VI-1 - AERODYNAMIC AND GEOMETRIC PROPERTIES OF MODEL 215

WING	
Area	838 ft^2
Mean Aerodynamic Chord	12.75 ft^2
Aspect Ratio (Geometric)	5.16
Section	NACA 64-221
HORIZONTAL TAIL	
Area	257 ft^2
Aspect Ratio	4.0
Moment Arm (Aft cg)	41.8 ft
Volume (aft cg)	1.01
Section	NACA 64-015
VERTICAL TAIL	
Area	161 ft^2
Aspect Ratio	1.0
Moment Arm (aft cg)	34.5 ft
Volume (aft cg)	.101
Section	NACA 64-015

WEIGHT-INERTIA

$W = 67{,}000$ lb

$I_{XX} = 983{,}400$ Slug - ft^2

$I_{YY} = 242{,}460$ Slug - ft^2

$I_{ZZ} = 1{,}128{,}450$ Slug - ft^2

$I_{XZ} = 12{,}100$ Slug - ft^2

due to camber which is a function of maximum thickness position as well as position and magnitude of maximum camber. Roughness merely decreases the energy of the boundary layer of thick airfoils, thus lowering maximum lift. Mach number effects are very severe on thick airfoils and maximum lift coefficient begins to drop at Mach 0.2. For thick cambered airfoils, the angle of attack for zero-lift varies with Mach Number, particularly above the critical Mach number. The calculation of $C_{L_{MAX}}$ and the C_L variation in the non-linear range are based on the methods of Reference VI-2 and the lift characteristics are given in Figure VI-5.

The Vertol method of drag build-up, as detailed in Reference VI-6 was used in this study to obtain the zero lift drag of the aircraft. The drag coefficient is defined as:

$$C_D = C_{D_{P_{MIN}}} + C_{D_P} + C_{D_I} + C_{D_M}$$

where $C_{D_{P_{MIN}}}$ = minimum parasite drag

C_{D_P} = parasite drag increase with lift

C_{D_I} = induced drag

C_{D_M} = drag due to compressibility

In cruise flight the total drag is due primarily to the $C_{D_{P_{MIN}}}$, since the drag due to lift is small at cruise lift coefficients, and the drag due to compressibility is reduced by selecting aircraft geometry to achieve that objective. The total parasite drag of each aircraft component is accounted for by the build-up of skin friction, three-dimensional effects, interference, and pressure drag due to flow separation. The results of these calculations are given in Table VI-2. The resultant equivalent drag area for the basic mission was then reduced to coefficient form, $C_{D_{P_{MIN}}}$, to which is added the drag due to lift. As cruise speed increases the effects of compressibility must be accounted for beginning at the critical Mach number. Above that speed boundary layer separation is caused by shock waves which results in a rapid drag rise. This effect on drag coefficient is provided for in the drag equation by the C_{D_M} term.

Wind tunnel test data of Reference VI-4 on the model shown in Figure VI-8 have been utilized to treat the full-span flap effect for the nominal flap angles of 0, 15, 30, 45 and 60 degrees, respectively. The test data provided excellent agreement with the lift and drag prediction at zero flap setting when corrected for the differences in aspect ratio and Reynolds number. The test lift curve slope after corrections was less than 2% higher than the calculated value, and the C_{L_O} intercept was 0.4 deg removed from the calculated value. The most noticeable difference between calculated

TABLE VI - 2

NUMBER
REV LTR
DATE:

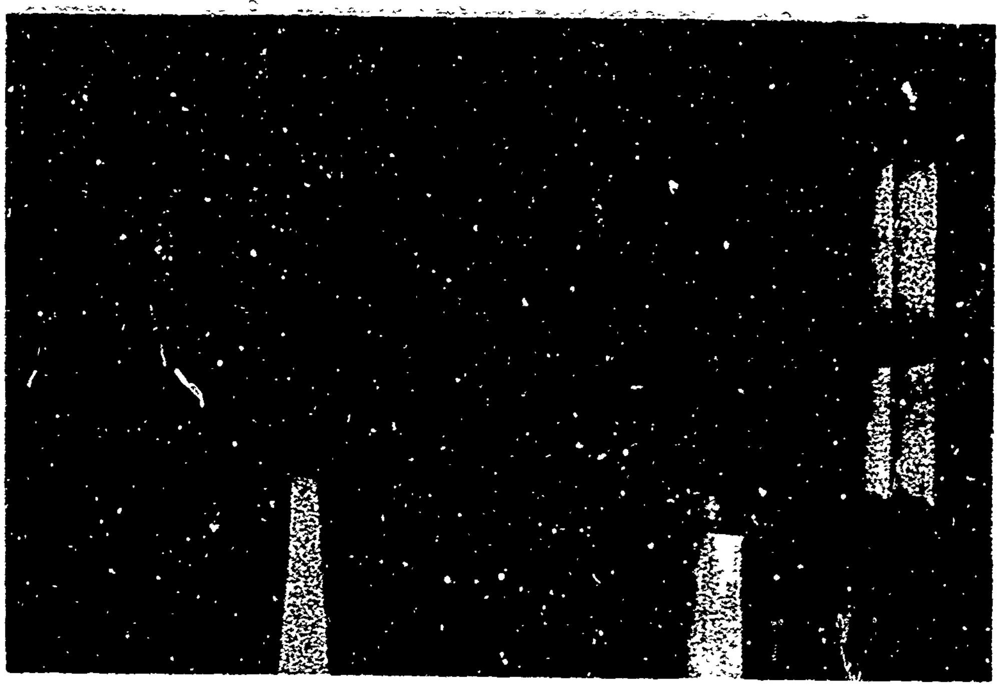
350 KT 10,000 STD. DAY
MINIMUM PARASITE DRAG BREAKDOWN

Configuration: USAF TILT ROTOR V/STOL AIRCRAFT MODEL 215

R_e/ft. = 2.9315×10^6 Drawing No.

COMPONENT	WETTED AREA	C_f	INCREMENT %	INCREMENT f_e	f_e (ft^2)
FUSELAGE	2040	.001832		3.7373	
3-Dimensional Effects				.3561	
Excrescences				.3000	
Canopy				.2140	
Afterbody				.7850	5.393
WING (21% t/c) (S_{REF}=838 FT^2)	1526	.002325		3.5480	
3-D Effects				1.802	
Excrescences				.2048	
Gaps [flaps, slats]				.4523	
[ailerons, spoilers]					
Body Interference				1.4510	7.4581
HORIZONTAL TAIL (15% t/c)	500	.002495		1.2475	
3-D Effects				.3710	
Excrescences & Taps				.1412	
Interference				.0395	1.7992
VERTICAL TAIL (15% t/c)	316.0	.002322		.705	
3-D Effects				.205	
Excrescences & Gaps				.079	
Interference				.041	1.045
INBOARD NACELLES					
3-D Effects					
Excrescences					
Interference					
Inlets					
Exhaust System					
OUTBOARD NACELLES (TILT ROTOR)	220.0	.002192		.4822	
3-D Effects per Nacelle				.0931	
Excrescences				.1184	
Interference				.0892	TOTAL
Inlets				.1354	
Exhaust System				.4075	2.6516
LANDING GEAR POD	120.0	.002332		.2798	
3-D Effects				.2062	
Excrescences				.1240	
Interference				.1240	.734
MISC. Roughness (6 % of $\Sigma C_f A_{WET}$)				.63	
MISC. Cooling				1.01	
MISC. Trim				.0704	
MISC. Air Conditioning					1.7104
TOTALS	4722.0	.002225			20.711

FORM 46284 (2/66)



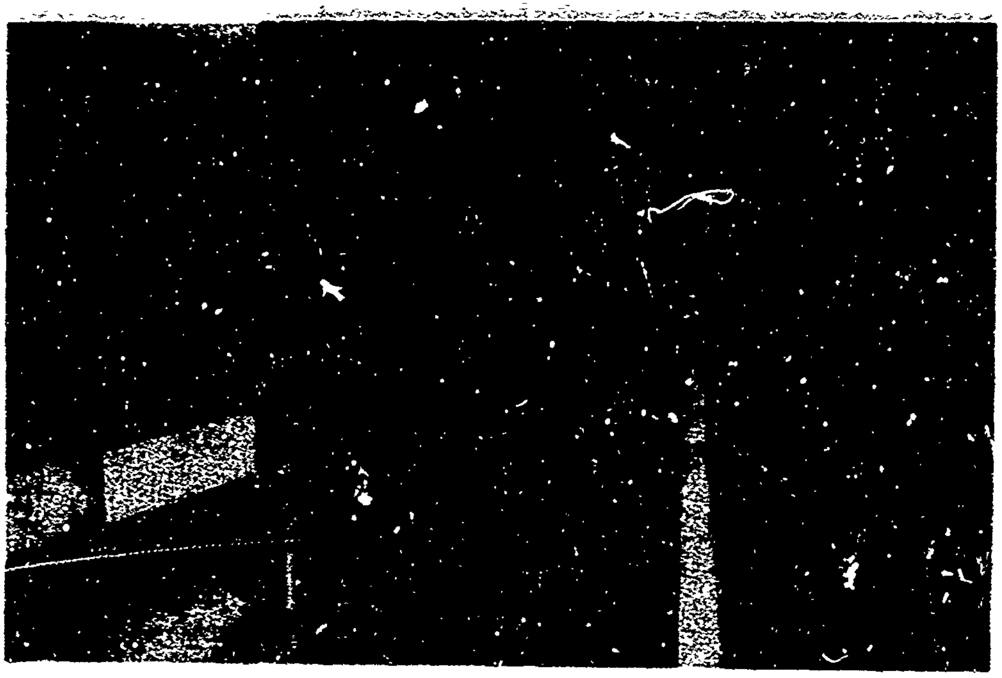


FIGURE VI-8

Model 150 Tilt Rotor Wind Tunnel Force Model and test results was that the test curve became non-linear at an angle of attack of 6°, compared with 11° for the calculated case. The no-flaps $C_{L_{MAX}}$ values, when reduced to the same flight conditions, were 1.179 for test and 1.178 for calculated results by the DATCOM method. Similar comparability was noted in the C_D vs α and C_L vs C_D curves.

The prediction of the effects of flap deflection on lift and drag is presented in Figure VI-7. For plain flaps approximating those on the Model 21. aircraft, the lift effects show only negligible differences but drag effects are in poor agreement at all flap angles. Therefore, the wind tunnel test data were used in determining the aerodynamic increments due to flaps. The slope of the lift curves for each flap setting were assumed in the calculations to be the same as for no-flaps case even though it is recognized that there is some change in slope due to changed wing geometry with flaps extended. The results used are similar to those obtainable by the method of Reference VI-7.

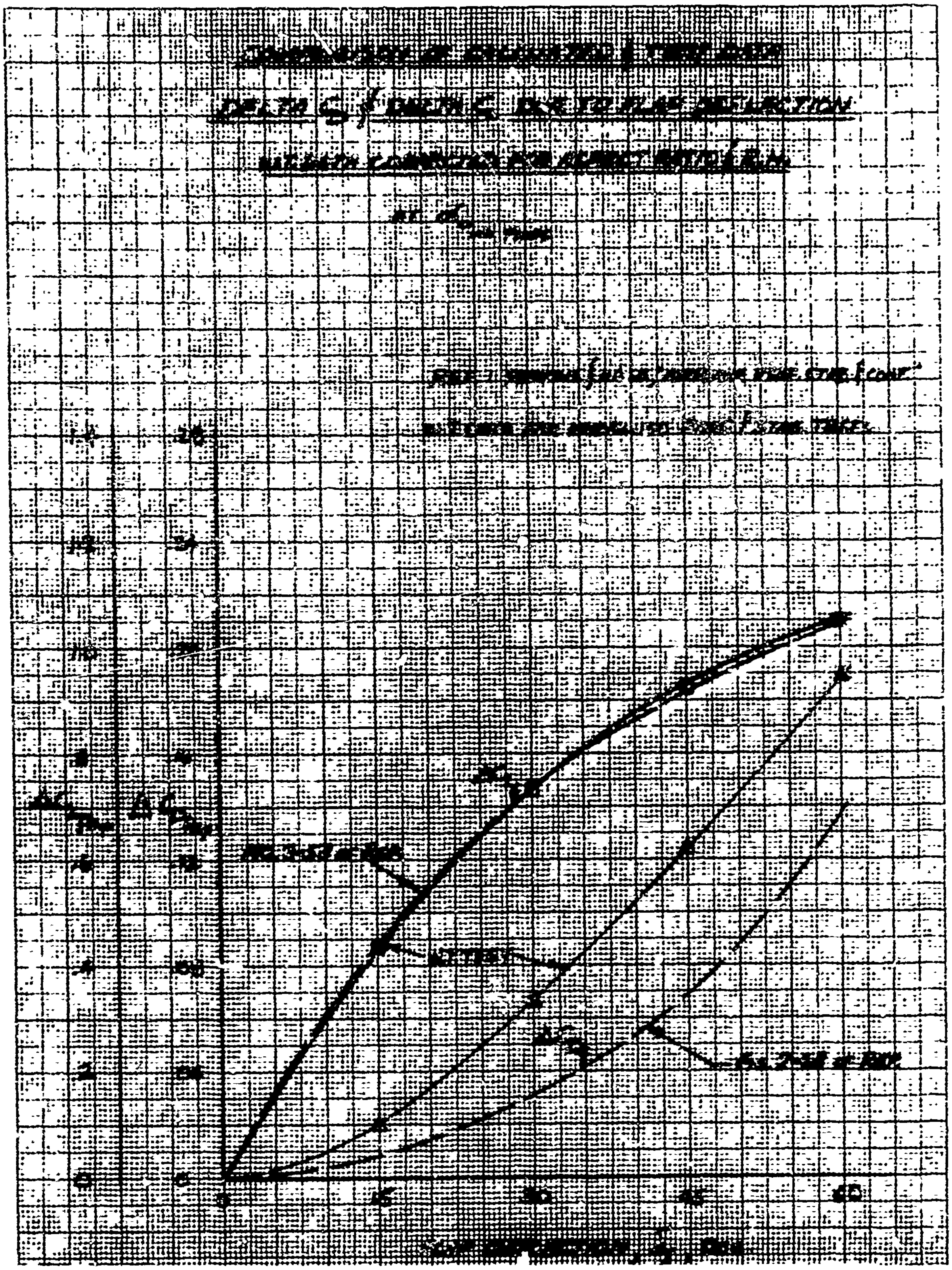


FIGURE VI-9 PREDICTION OF THE EFFECT OF FLAPS SHOWS SAME LIFT AND LESS DRAG THAN WIND TUNNEL MODEL TEST

5. STATIC STABILITY IN AIRPLANE FLIGHT

The LARMS load alleviation flight control system has a large effect on static stability since it cancels all of the static effects of the rotors. To provide for the possibility of this system being inoperative, the empennage were sized to provide neutral static angle of attack and directional stability at 1.15 times the 30 degree flap stall speed at the most aft flight center of gravity location including the destabilizing effects of the rotors. This speed corresponds to the minimum, rotors fully converted, flight velocity during transition. The horizontal tail area and tail volume are 257 square feet and 1.01 respectively (referred to the aft c.g.). The vertical tail area and tail volume are 161 square feet and 0.101 respectively.

The horizontal tail is located on top of the vertical tail to minimize the wing downwash and dynamic pressure loss effects which are destabilizing. Also, the high horizontal tail acts as an endplate on the vertical tail to increase the vertical tail effective aspect ratio.

The angle of attack stability of the aircraft with this tail size and without the destabilizing effects of the rotors is shown in Figures VI-10 and VI-11 for the transition and cruise mode respectively. The static margin this stability produces is slightly stable if the rotor stabilization system is inoperative as shown in Figure VI-12. With the rotor stabilization system operating the static margin at the aft cg limit is greater than 25 percent at all flight speeds in the airplane mode.

The directional stability is also near neutral with the LARMS rotor stabilization system inoperative with this tail size. As shown in Figure VI-13 a very high level of directional stability is provided when this system in operating such as to cancel the rotor effects.

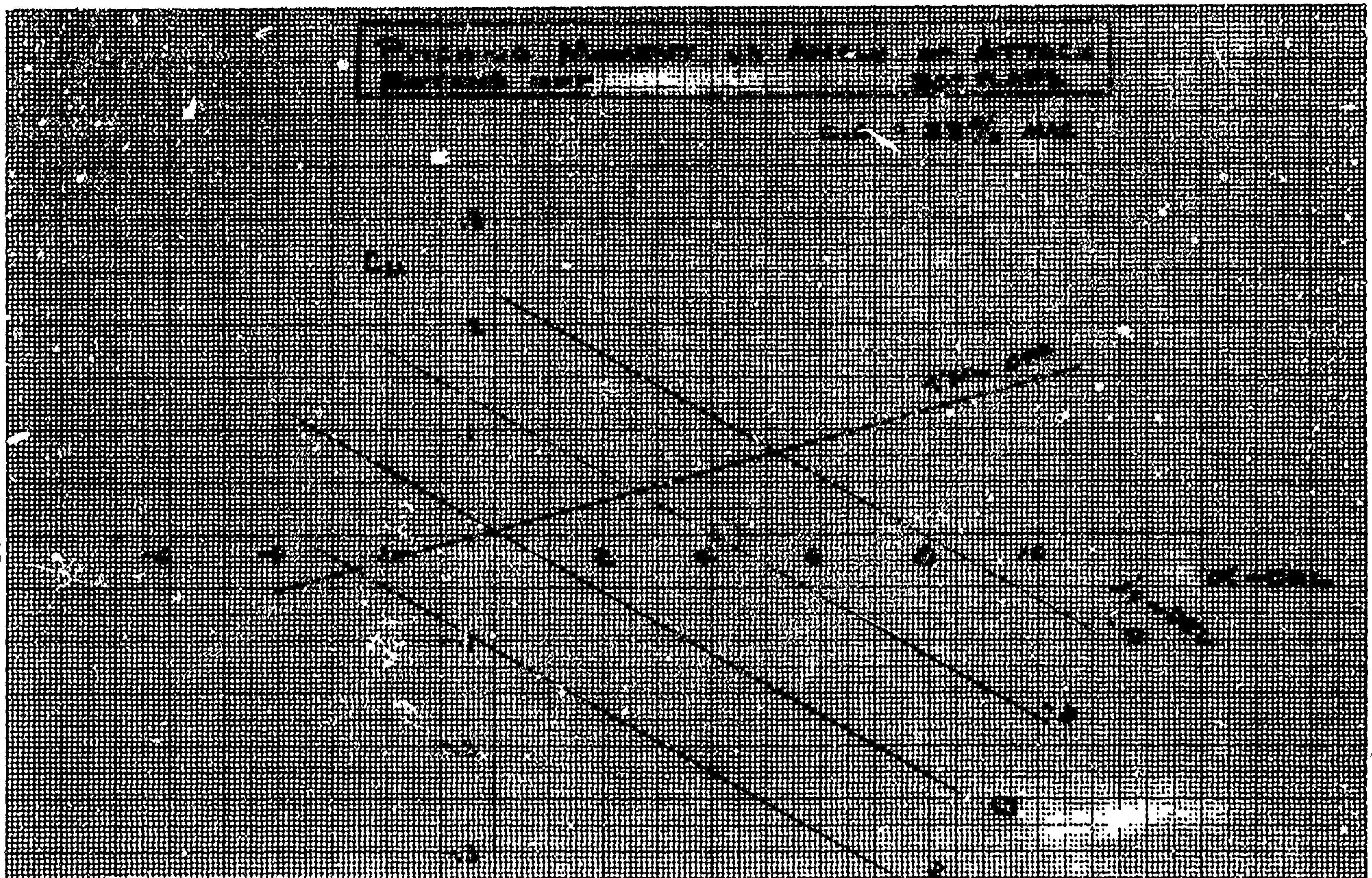


FIGURE VI-10 ANGLE OF ATTACK STABILITY OF MODEL 215 WITH FLAPS DOWN AND ROTORS OFF

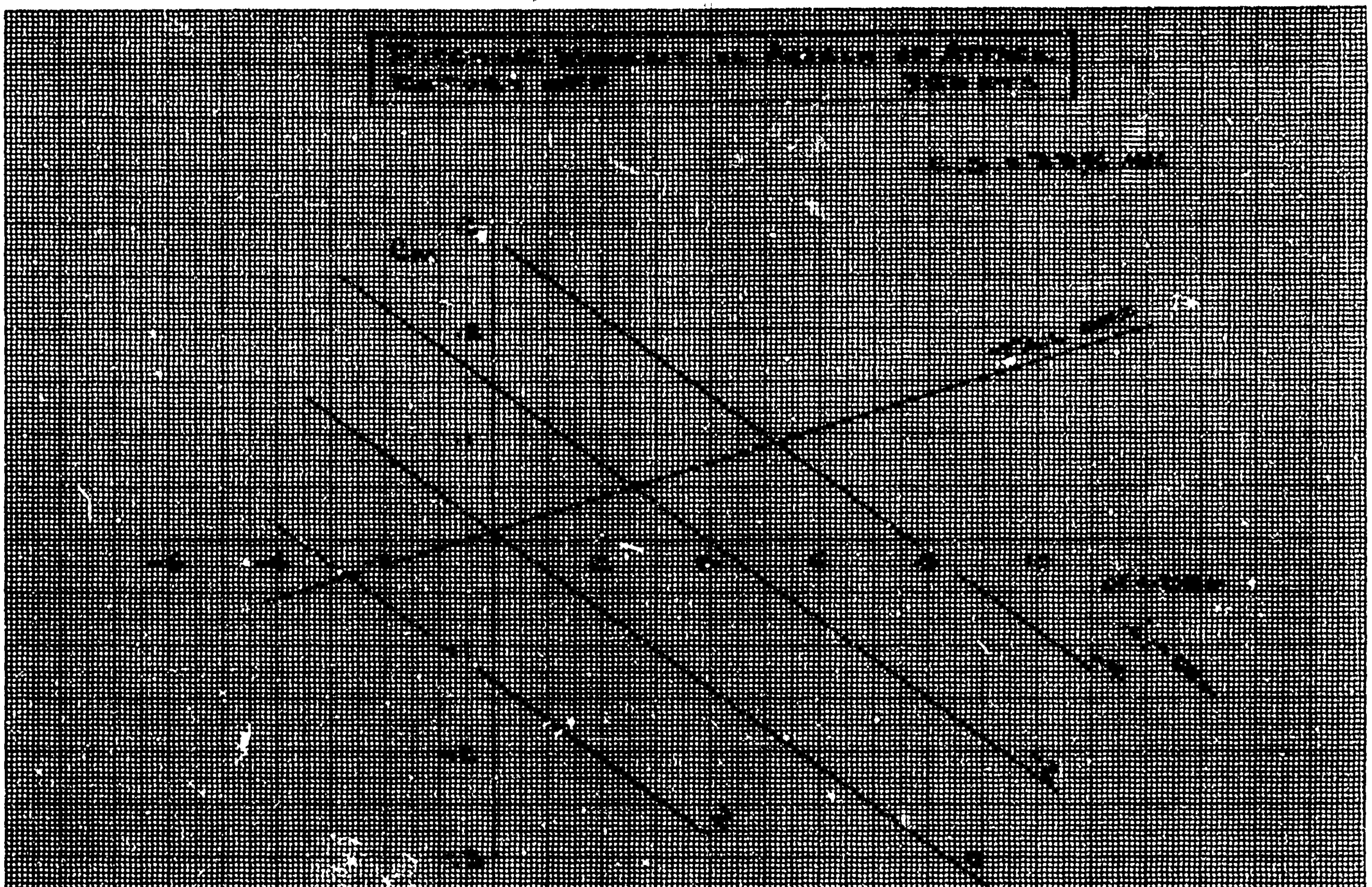


FIGURE VI-11 ANGLE OF ATTACK STABILITY OF MODEL 215 IN CRUISE

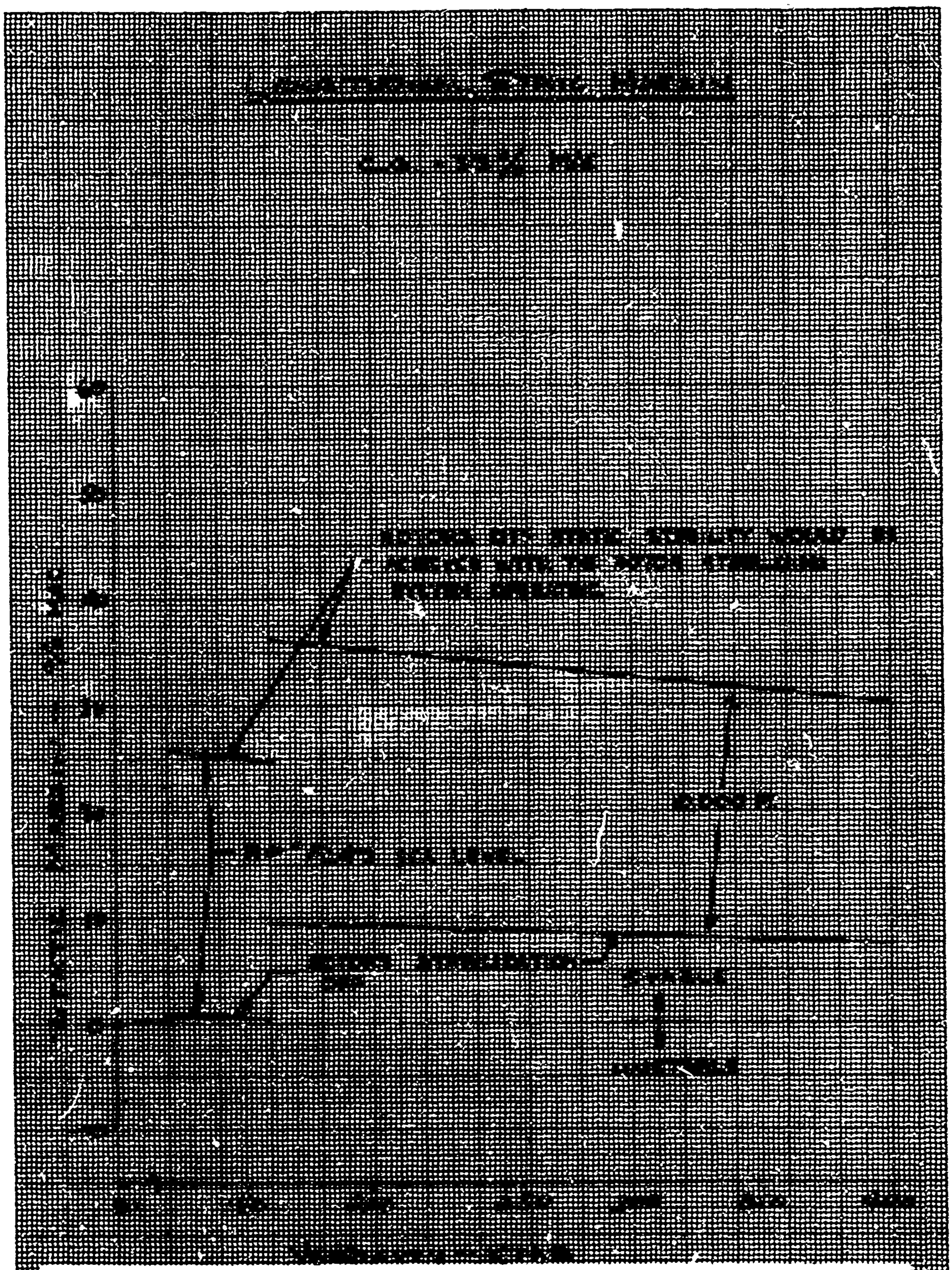


FIGURE VI-12 HORIZONTAL TAIL SIZING PRODUCES ALMOST NEUTRAL STABILITY AT UPPER END OF TRANSITION WITH ROTOR STABILIZATION OFF

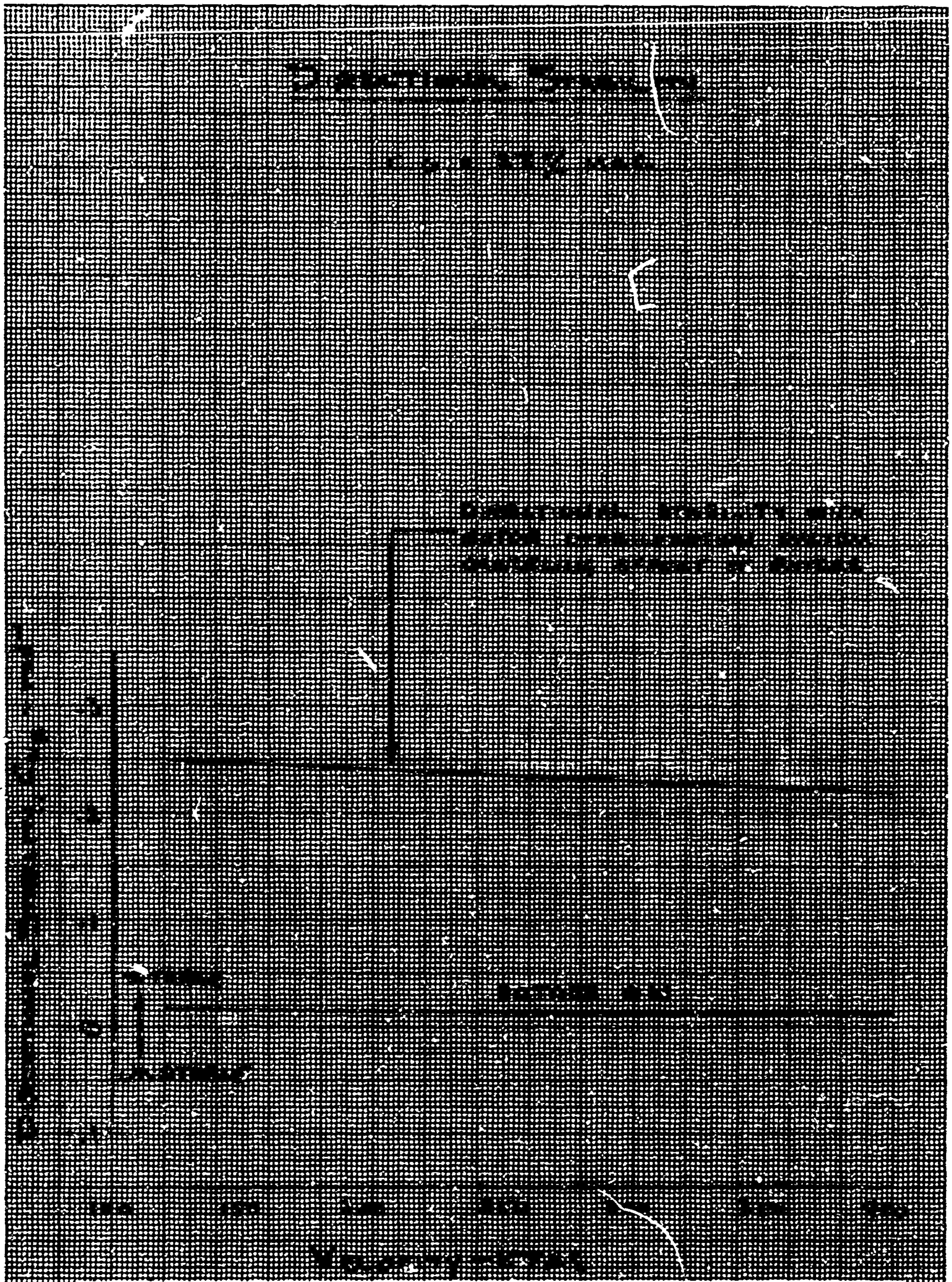


FIGURE VI-13 DIRECTIONAL STATIC STABILITY OF MODEL 215

6. STABILITY DERIVATIVES IN AIRPLANE MODE

The static and dynamic stability derivatives used in the following dynamic stability analysis are summarized in Tables VI-3 and VI-4. These derivatives are from the previously given static stability analysis and/or were obtained from combined rotor and airplane theory and data.

The rotor thrust variation with velocity was estimated from Reference VI-8 which utilizes an explicit vortex influence technique; the EVIT Program. The rotor normal force variation with angle of attack was estimated from Reference VI-2 methodology which is based on rigid propeller test data. To properly account for the flapping phase relationship on the rotor pitching moment variations with angle of attack the L-02 aeroelastic rotor program, Reference VII-9, was utilized which provides a complete aeroelastic representation of the rotors. This program shows good correlation with previous Vertol rotor test data. The rotor-airframe and airframe-rotor interference effects were estimated from Reference VI-2. Conventional methodology from References VI-2 and VI-5 was utilized to predict the rotors off stability derivatives. This procedure involved a buildup from two dimensional airfoil data and correcting for aspect ratio, compressibility effects, interference, etc. This procedure

TABLE VI-3. LONGITUDINAL STABILITY DERIVATIVES FOR MODEL 215 AIRCRAFT

DERIVATIVE W = 67,000 LB CG = .33$\bar{c}$	V = 135 KT SEA LEVEL		V = 180 KT H = 10,000 FT		V = 350 KT H = 10,000 FT	
	ROTORS OFF	ROTORS ON	ROTORS OFF	ROTORS ON	ROTORS OFF	ROTORS ON
$\frac{dF_x}{mdu}$ - sec^{-1}	-.0292	-.3080	.0072	-.1860	-.0130	-.1571
$\frac{dF_z}{mdu}$ - sec^{-1}	-.2143	-.2415	-.1898	-.2235	-.1216	-.1216
$\frac{dM_y}{I_y du}$ - ft^{-1} sec^{-1}	-.0004	.0104	-.0024	.0050	-.0018	.0024
$\frac{dF_x}{mdw}$ - sec^{-1}	.1201	.1201	.0280	.0280	-.0429	-.0429
$\frac{dF_z}{mdw}$ - sec^{-1}	-.6151	-.7390	-.5843	-.7136	-1.249	-1.571
$\frac{dM_y}{I_y dW}$ - ft^{-1} sec^{-1}	-.0170	-.00055	-.0228	-.0073	-.0429	-.0136
$\frac{dM_y}{I_y d}$[illegible] - rad^{-1} sec^{-1}	-1.50	-1.66	-1.52	-1.66	-3.25	-3.51
$\frac{dM_y}{I_y d\dot{\alpha}}$ - rad^{-1} sec^{-1}	-.44	-.48	-.41	-.45	-1.06	-1.15

TABLE VI-4. LATERAL - DIRECTIONAL STABILITY DERIVATIVES FOR MODEL 215 AIRCRAFT

DERIVATIVE		V = 135 KT SEA LEVEL ROTORS OFF	V = 135 KT SEA LEVEL ROTORS ON	V = 350 KT H = 10,000 FT ROTORS OFF	V = 350 KT H = 10,000 FT ROTORS ON
$\frac{dF_y}{m dv}$	- sec^{-1}	-.099	-.202	-.194	-.478
$\frac{d\mathcal{N}}{I_z dv}$	- ft^{-1} sec^{-1}	.0033	.00023	.0057	.00033
$\frac{d\mathcal{L}}{I_x dv}$	- ft^{-1} sec^{-1}	-.00148	-.00154	-.0029	-.0034
$\frac{dF_y}{m d\dot{\phi}}$	- ft/rad sec	-.401	-.401	-.808	-.808
$\frac{d\mathcal{N}}{I_z d\dot{\phi}}$	- rad^{-1} sec^{-1}	.0859	.0859	.0687	.0687
$\frac{d\mathcal{L}}{I_x d\dot{\phi}}$	- rad^{-1} sec^{-1}	-.171	-.398	-.406	-1.04
$\frac{dF_y}{m d\dot{\psi}}$	- ft/rad sec	1.79	-2.66	-3.59	-6.01
$\frac{d\mathcal{N}}{I_z d\dot{\psi}}$	- rad^{-1} sec^{-1}	-.143	-.185	-.232	-.287
$\frac{d\mathcal{L}}{I_x d\dot{\psi}}$	- rad^{-1} sec^{-1}	.159	.161	.367	.372

was performed on the similar configuration from Reference VI-4 and shows good correlation with the wind tunnel test data.

7. DYNAMIC STABILITY IN AIRPLANE MODE

A preliminary evaluation of the aircraft shows a minimum requirement for stability augmentation and essentially no dependence on the rotor stabilization system (LARMS) for adequate stick-fixed flying qualities. Analytical results for the operational flight envelope up to 20,000 feet and above V_{CON} are given for the aircraft normal state and with the rotor stabilization system failed. This kind of failure is considered as remotely possible so level three (3) flying qualities are desired and are shown to be easily achieved after this failure.

This analysis is based on the stability derivatives given in the previous section; airplane dynamic derivatives are from DATCOM or Reference VI-5 and rotor derivatives from standard rotor analysis, Reference VI-9. The parameters of the aircraft used for this analysis are summarized in Table VI-1. Presently, the control system of the aircraft and its rotor system has not been adequately defined for detailed stability analysis. As discussed in Section 3.0, it is anticipated that the primary control of the aircraft will be provided by the rotor control system with the proper feel - feedbacks or some advanced pilot control system is assumed to be provided in this aircraft so that there will be no deficiency in flying qualities resulting from this control system.

Longitudinal

a. Short Period: The short period motion both with and without rotor effects is well within the spec for Category B. Data for the longitudinal short period frequency and acceleration sensitivity of the aircraft are shown in Figure VI-14 for the aircraft with the rotor effect cancelled by the LARMS control system. If this system was inoperative, slightly better short period flying qualities would result as shown in Figure VI-15. In either case, speeds from 180 to 350 knots, altitudes to 20,000 feet and C.G. positions over the allowable range produce adequate flying qualities This parameter is also excellent at the upper transition speed of 135 knots with the flaps down at the aft C.G..

The damping of the short period mode of the aircraft with the aircraft in its normal state is shown in Figure VI-16 to be excellent. Figure VI-17 shows that complete failure of the rotor stabilization (LARMS) system reduces the period of the short period mode but greatly increases the damping of this mode. This shows that the LARMS

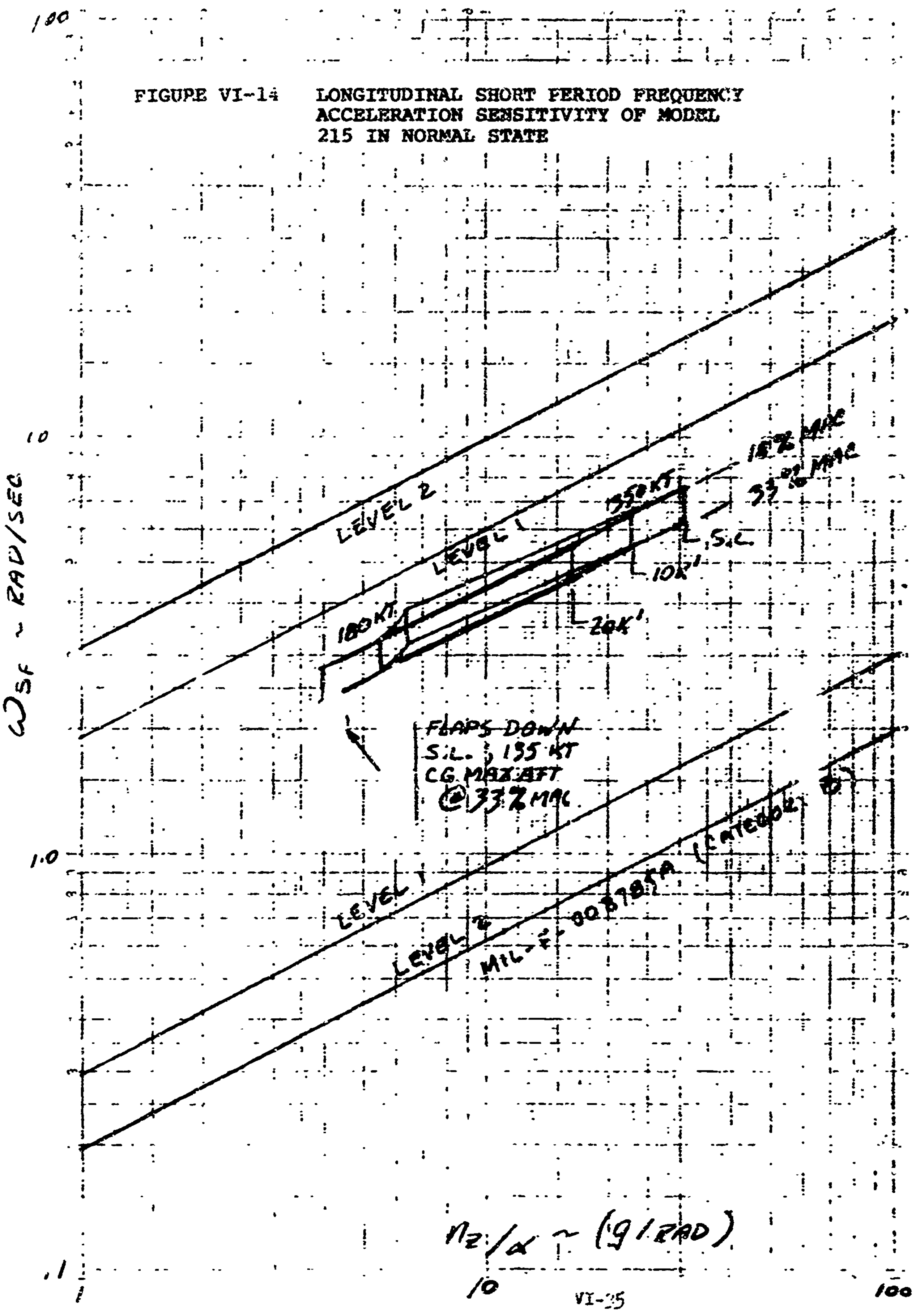


FIGURE VI-14 LONGITUDINAL SHORT PERIOD FREQUENCY ACCELERATION SENSITIVITY OF MODEL 215 IN NORMAL STATE

FIGURE VI-15 LONGITUDINAL SHORT PERIOD FREQUENCY AND ACCELERATION SENSITIVITY OF MODEL 215 AFTER REMOTELY PROBABLE COMPLETE FAILURE OF ROTOR STABILIZATION SYSTEM

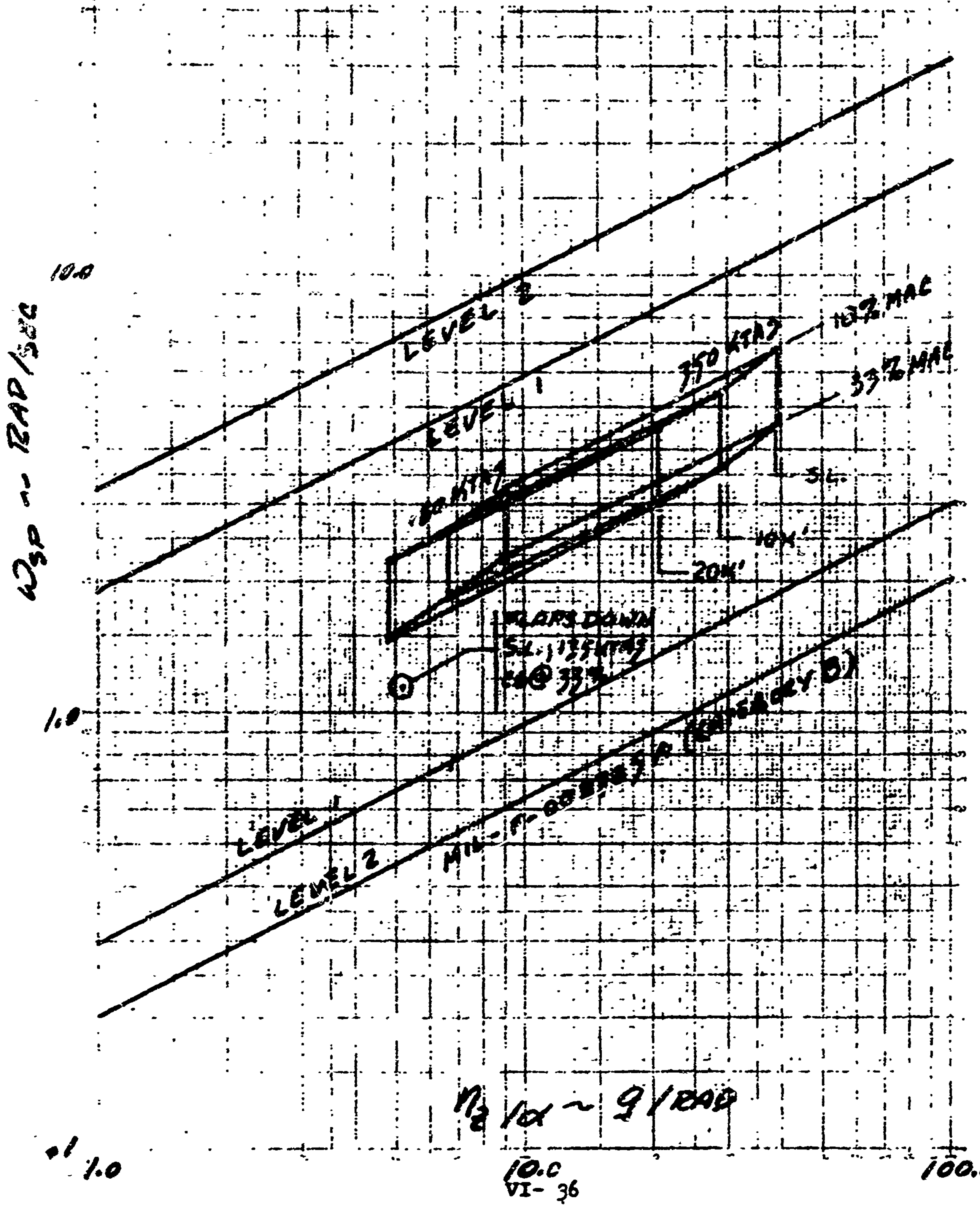


FIGURE VI-16 LONGITUDINAL SHORT PERIOD DAMPING RATIO OF MODEL 215 IN NORMAL STATE

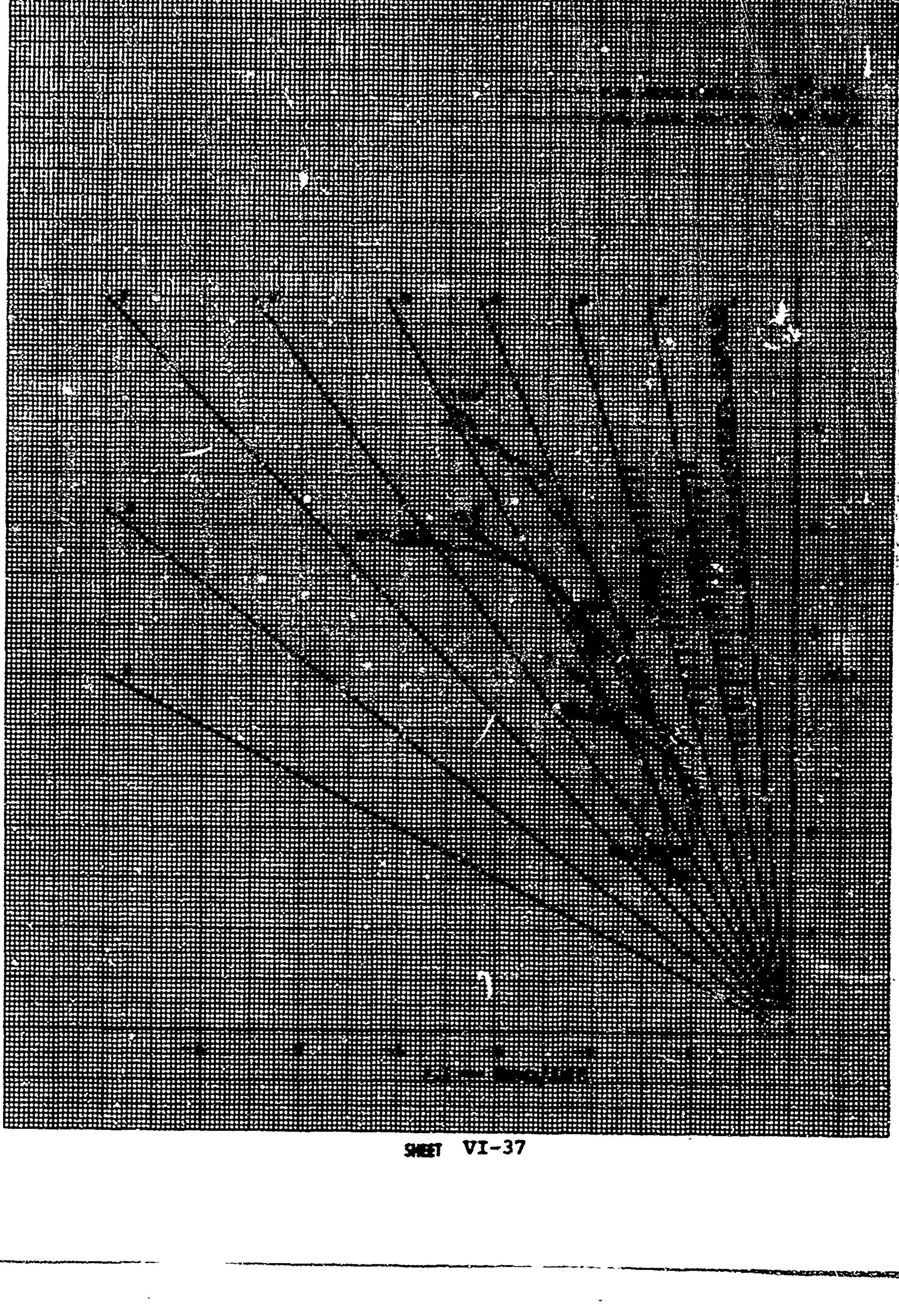


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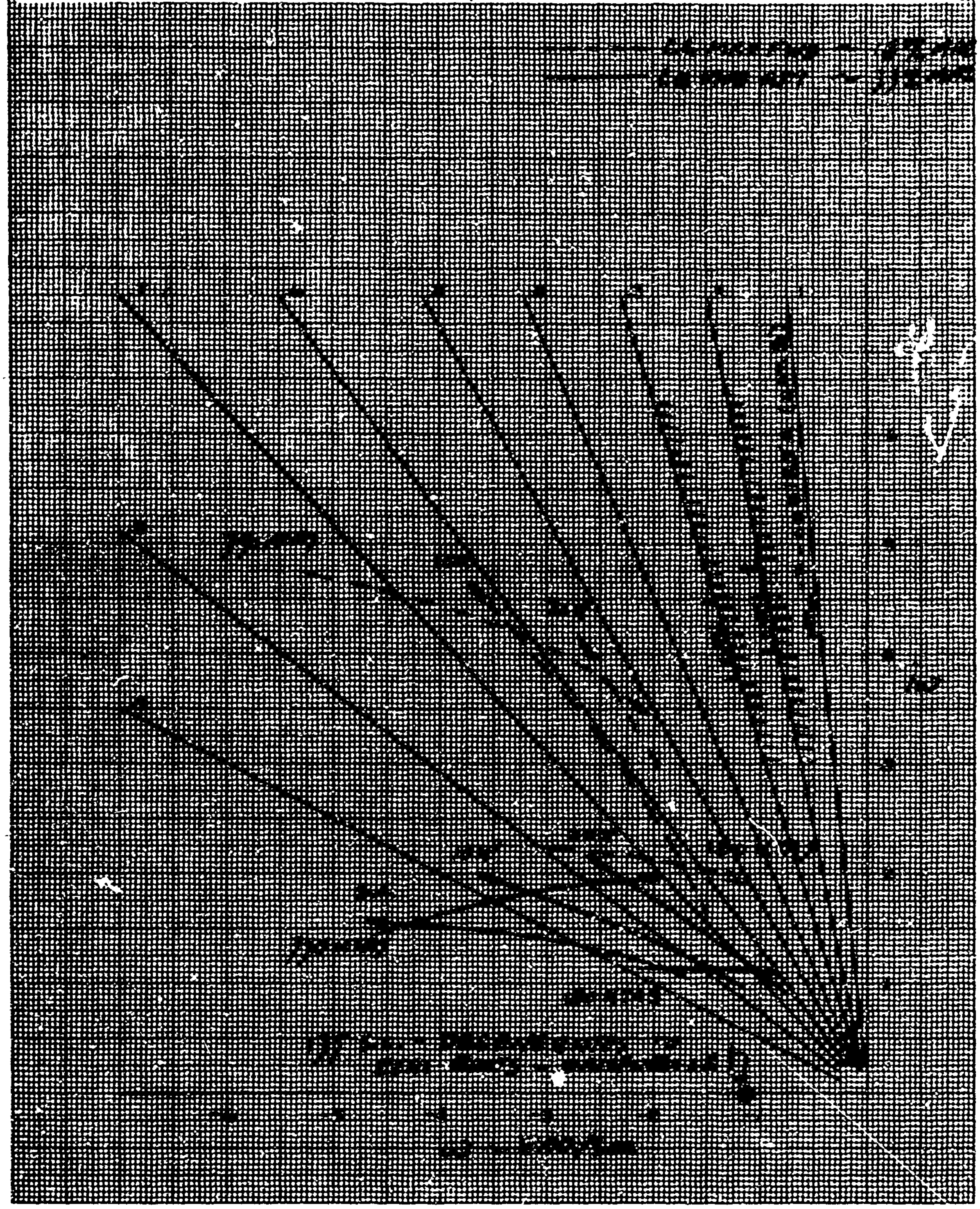
FIGURE VI-17 LONGITUDINAL SHORT PERIOD DAMPING RATIO OF AIRCRAFT AFTER REMOTELY PROBABLE COMPLETE FAILURE OF ROTOR STABILIZATION SYSTEM (LARMS)



system will significantly improve the gust response characteristics of the aircraft.

b. Phugoid: Analysis of the controls fixed phugoid mode shows that the horizontal gust alleviation function of the IARMS system should not completely cancel the sensitivity of the rotors to velocity perturbations. As shown in Figure VI-18, the aircraft with the gust alleviation system inoperative has a highly damped phugoid indicative of gust sensitivity. If the gust alleviation system were to completely cancel the effects of the rotors, a slightly unstable phugoid results. This system will be developed to produce the minimum phugoid damping required to produce "level 1" flying qualities.

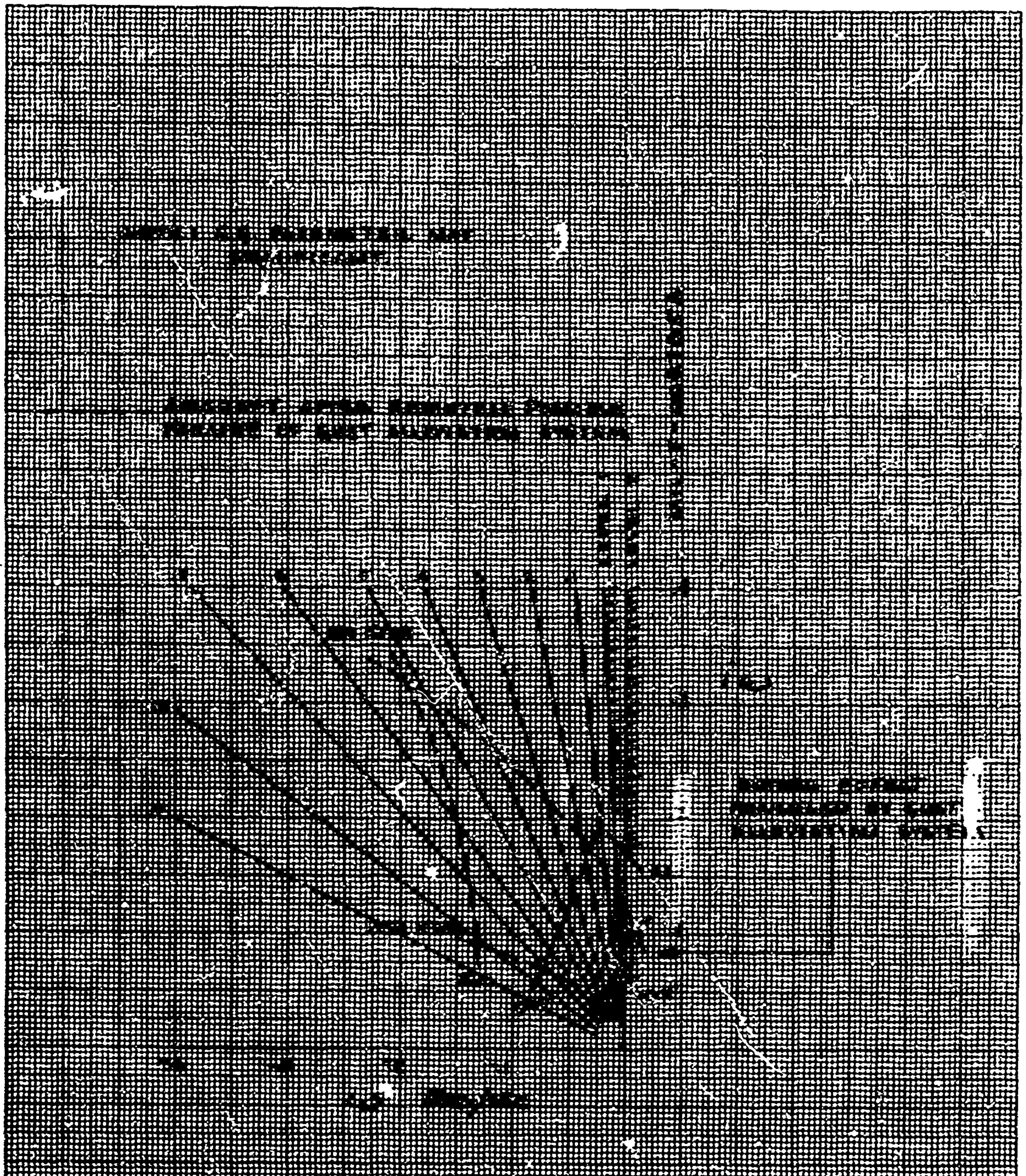


FIGURE VI-18 LONGITUDINAL PHUGOID DAMPING WILL REQUIRE GUST ALLEVIATION SYSTEM IN ROTOR CONTROLS

Lateral

(a) Dutch Roll: The dutch roll characteristics are deficient in comparison with MIL-F-008785 and yaw rate feedback to the directional controls is required. This is shown uncorrected in Figure V -19. Again it is observed that in case of a failure of the rotor stabilization system, the characteristics are manageable. The control authority requirements to damp the rotor stabilized configuration are small.

(b) Roll Subsidence: From this preliminary assessment, the aircraft is deficient in roll damping as a result of a high roll inertia as compared to the wing span. As shown in Figure VI-20, the roll damping of the unaugmented airplane does not satisfy the Level 1 specification except for high speed - low altitude flight with the rotor stabilizing system inoperative. When the damping provided by the rotor is removed by the rotor stabilization, only Level 2 damping of the aircraft is provided. This problem is expected to be solved by close attention to providing rotor nacelles which increase the effective wing span. The beneficial influence of the rotor nacelles has been neglected in this initial assessment of the problem. Addition of small wing panels to the outboard sides of the rotor nacelles would provide adequate damping if the nacelles can not be made to provide adequate effective span.

(c) Spiral Divergence: As indicated by the tail sizing philospohy, neutral stability power on will produce an unstable spiral with the rotors stabilized. This is to be expected due to the large effect of the unaugmented rotors in yaw. As shown in Figure VI-21, the rotor stabilization should allow a small destabilizing effect with sideslip to prevent a spiral divergence.

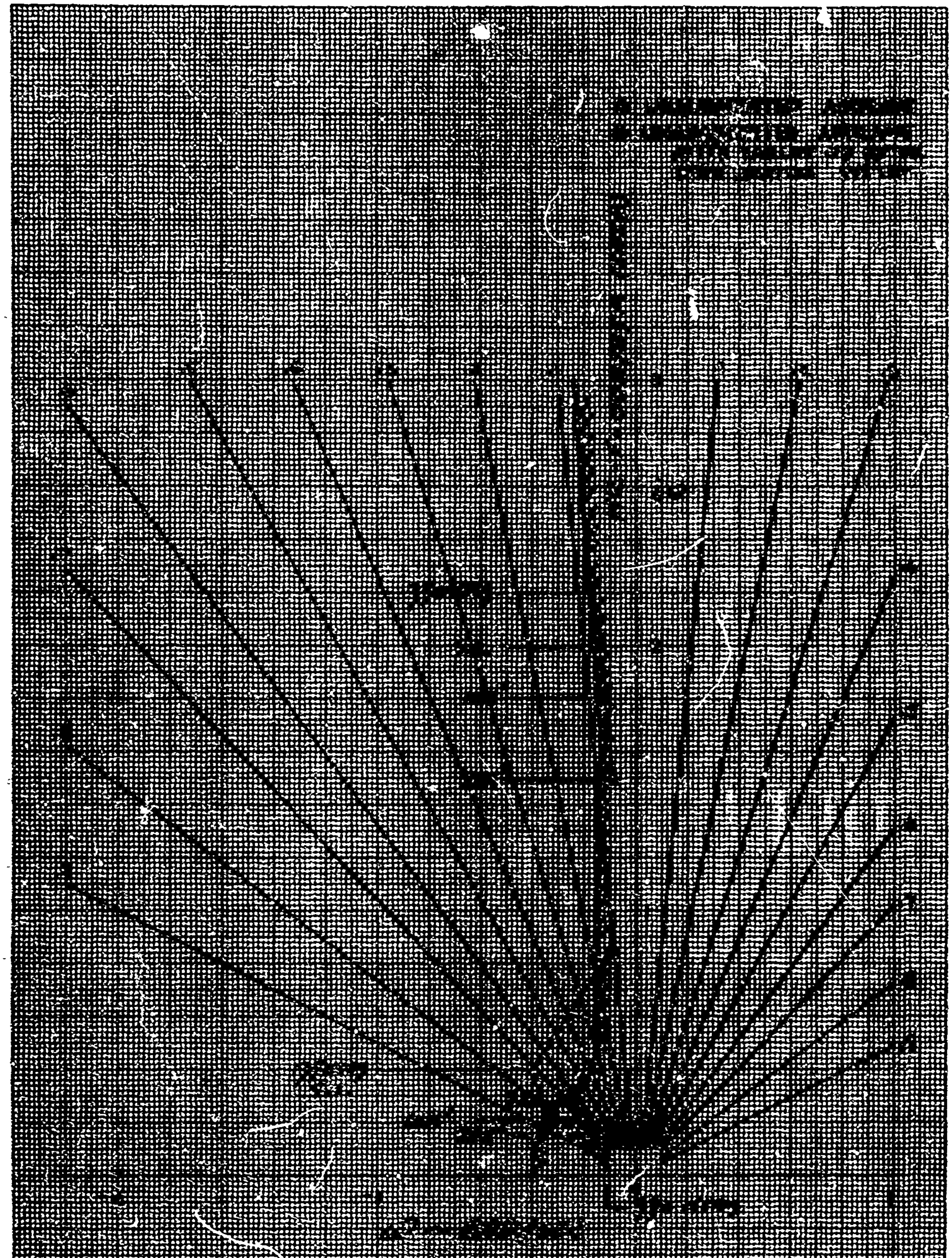
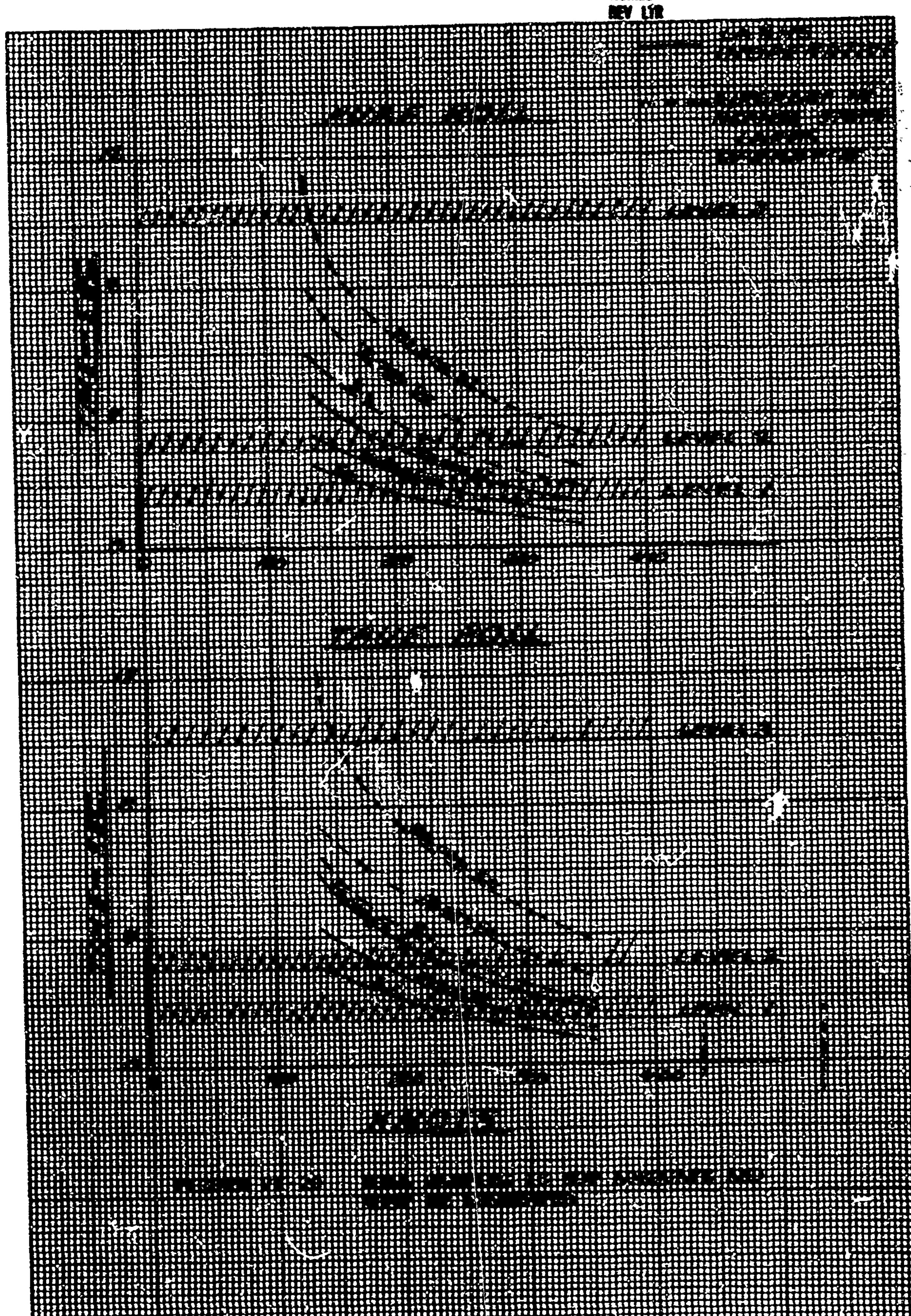


FIGURE VI-19 DUTCH-ROLL FREQUENCY AND DAMPING OF UNAUGMENTED MODEL 215 SHOWS NEED FOR YAW RATE AUGMENTATION



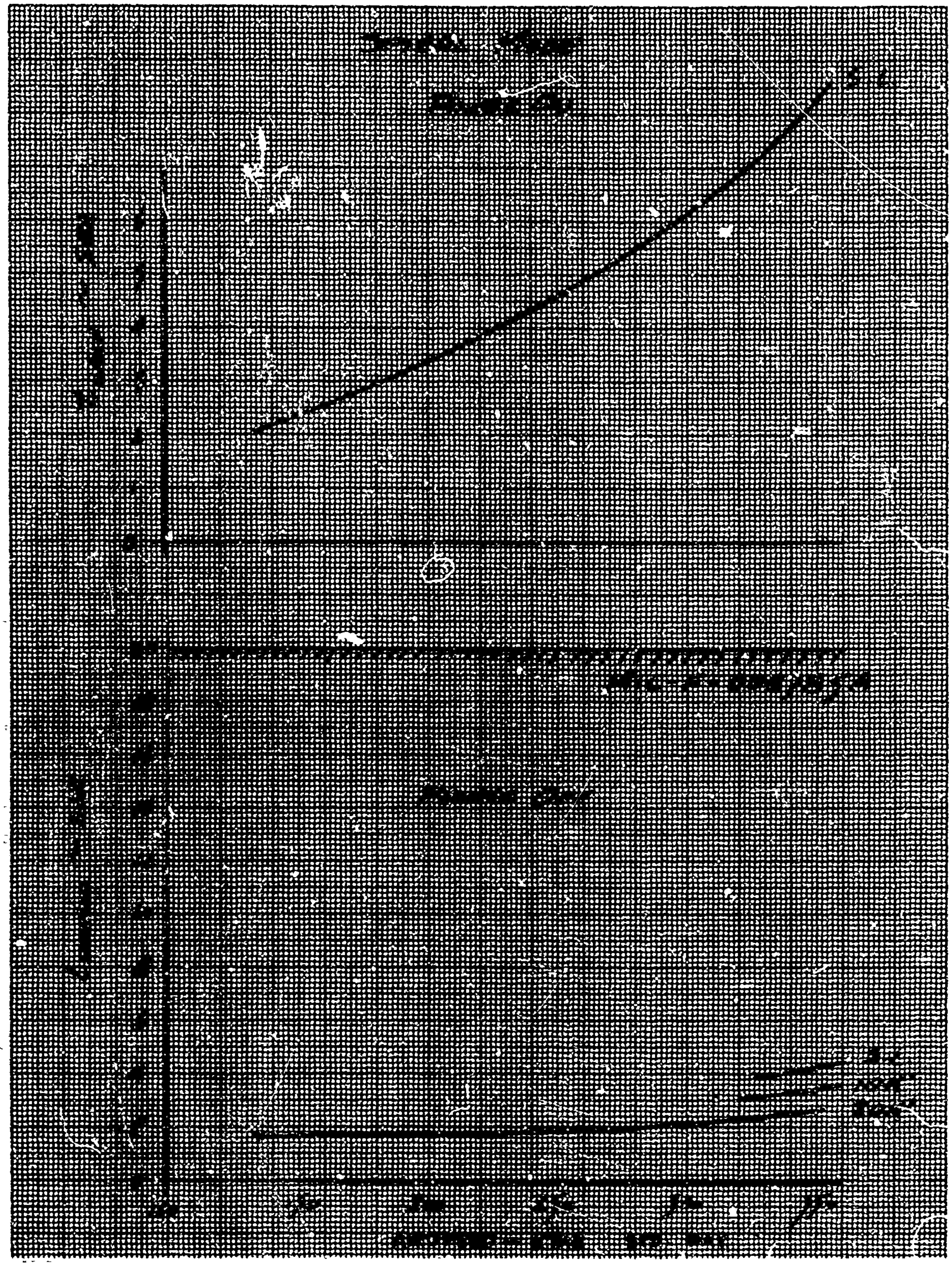


FIGURE VI-21 SPIRAL STABILITY SHOWS EXCESSIVE DIRECTIONAL STABILITY OF MODEL 215 WITH ROTOR STABILIZING SYSTEM

8. GUST RESPONSE

The tilt rotor aircraft will have acceptable gust response characteristics due to the provision of rotor cyclic pitch feedback through the load-alleviation system and rotor collective pitch feedback of horizontal nacelle acceleration. This system is expected to be able to keep the cabin response due to horizontal, lateral and vertical discrete (1-Cosine) gusts up to 20 ft/sec amplitude less than the following values:

0.1 g's vertically

0.05g's laterally

0.05g's horizontal

Additionally, this level of maximum gust response amplitude will also be shown using temperal gust variations such as those given by the statistical models of AFFDL-TR-68-85. These gust levels do not require any additional authority of the cyclic feedback system or the collective pitch control. Response of these control systems also appear to be adequate.

9. VIBRATION

The tilt rotor aircraft will have vibration levels within the proposed Boeing criteria for occupied areas of 0.05 g's at the number of blades per rev. frequency and well within the MIL-H8501A requirement of 0.15 g's. This goal will be achieved by tuning the wing vertical bending stiffness to alternate the number of blades per rev. frequency. The present preliminary design shows that the wing stiffness can be reduced without compromising the whirl flutter or air/ground resonance stability. Necessity for such alternation only arises during transition with very small oscillating rotor forces being predicted for cruise flight in the airplane mode.

Vibration in transition presently can not be analytically predicted with confidence but statistical prediction techniques are available for helicopter preliminary design. It can be shown that vertical vibration varies as the product of the square of the rotor inplane advance ratio and the thrust coefficient-solidity ratio. The tilt rotor aircraft only flies edgewise in helicopter flight and in transition. Use of the helicopter statistics and a typical nacelle incidence schedule for transition produces the vibration prediction shown in Figure VI-22. A maximum vibration level of 0.11 g's is shown to occur at about 100 knots. This vibration level should be alleviated since a speed of 100 knots is likely to be used for low speed loiter and search operations in the rescue mission. Alleviation may be achieved by absorber, force balancers or by selection of the wing stiffness to attenuate the rotor forces felt at the fuselage.

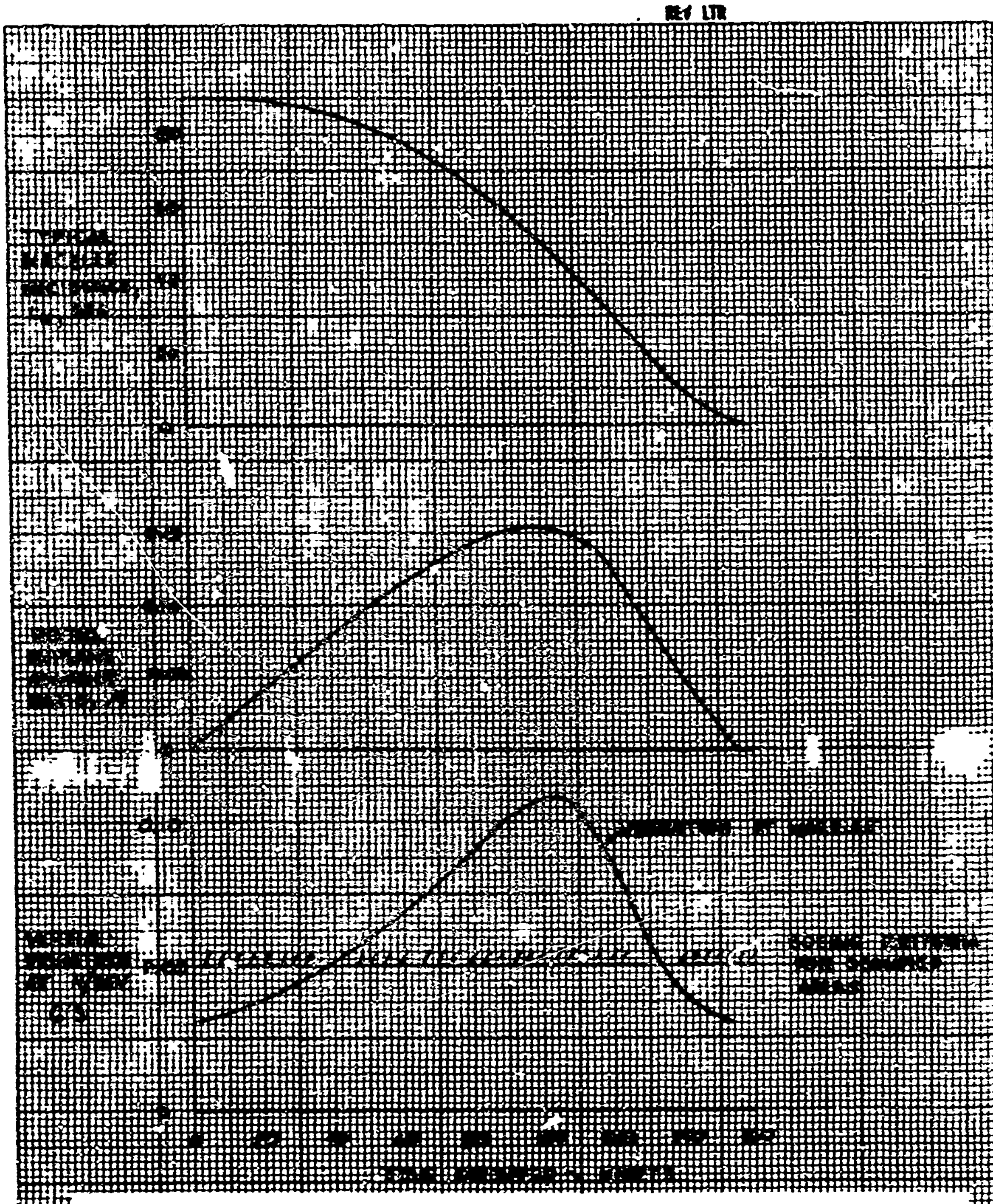


FIGURE VI-22 ESTIMATED CABIN VIBRATION LEVEL DURING UN-ACCELERATED TRANSITION IN LEVEL FLIGHT

SECTION VII

STRUCTURES

1. SUMMARY

This section contains criteria for use during Phase II, the structural design of the prop/rotor aircraft rotor blades, hub, wing, nacelle structure and transmissions. Limit load and fatigue conditions are included. Specifications MIL-A-8860 and MIL-S-8698 have been used to guide the selection of conditions and only those which are generally critical are to be considered for preliminary design purposes.

2. APPLICABLE SPECIFICATIONS

The structural design criteria shall be in general accord with the following military specifications with consideration given to that required for preliminary design:

a. MIL-A-8860, "General Specification for Airplane Strength and Rigidity".

b. MIL-S-8698, "Structural Design Requirements, Helicopters".

3. FLIGHT MODE DEFINITION

The flight modes for the vehicle are defined as:

a. Helicopter Flight: Lift is provided only by the rotor and airspeeds are less than 35 knots in any direction.

b. Transition Flight: Lift is provided by the rotor and the wing. This regime starts at 35 knots and ends at V_{CON}.

c. Airplane Flight: Lift is provided only by the wing. The regime starts at V_{CON} and is limited at V_L.

d. V_{CON} is the airspeed at which $n_z = 1.2$ can be achieved with the flaps retracted.

4. BASIC DESIGN PARAMTERS

The basic design parameters for the three flight modes are listed in Table VII-1.

5. FACTOR OF SAFETY

The yield factor of safety shall be 1.0. The ultimate factor of safety shall be 1.5.

6. TORQUE FACTOR

The limit torque factor shall be 1.5.

7. DESIGN SPEED

A. For helicopter flight, the maximum forward, sideward and rearward speed shall be 35 knots.

B. For transition flight, the design speed for limit load conditions shall be the minimum speed indicated on the V-n diagram for which a 3.0 limit load factor applys.

C. For airplane flight, the following speeds apply:

1. Maximum level flight speed V_H equal to 360 knots (transmission torque limit) at sea level.

2. The limit speed V_L shall be 450 knots (1.25 V_H) at sea level.

TABLE VII-1

BASIC DESIGN PARAMETERS

PARAMETER	DESIGN VALUE
HELICOPTER FLIGHT	
Basic Design Gross Weight	67,000 lb.
Minimum Flying Gross Weight	47,798 lb.
Most Aft C.G. Position	F.S. 421.6 in.
Most Forward C.G. Position	F.S. 398.7 in.
Limit Load Factor at Basic Design Gross Weight (N_z)	2.5, -1.0
Limit Landing Sinking Speed at Basic Design Gross Weight	12.0 fps (See Note 1)
Normal Rotor Speed, Power on	295
Rotor Speed Limit Factor	1.25
Nacelle Axle	F.S. 410
TRANSITION FLIGHT	
Basic Design Gross Weight	67,000 lb.
Maximum Design Gross Weight	74,000 lb.
Limit Load Factor at Basic Design Gross Weight (N_z)	3.0, -1.0
Normal Rotor Speed, Power on	295 RPM
Rotor Speed Limit Factor	1.25
AIRPLANE FLIGHT	
Basic Design Gross Weight	67,000 lb.
Maximum Design Gross Weight	74,000 lb.
Minimum Flying Gross Weight	47,798 lb.
Most Aft C.G. Position	F.S. 402.5 in.
Most Forward C.G. Position	F.S. 379.5 in.
Limit Load Factor at Basic Design Gross Weight	3.0, -1.0
Normal Rotor Speed	207 RPM

NOTE 1: The limit landing load factor shall be based upon a sinking speed of 12 fps and rotor lift equal to two-thirds of the basic design gross weight.

3. The maximum speed for a 66 fps gust V_G shall be 260 knots (S.L.) for the basic design gross weight and 240 knots (S.L.) for the minimum flying gross weight, $V_G = \sqrt{n}V_S$ where n is the maximum gust load factor determined at V_H and V_S is the stalling speed for level flight at sea level in the basic configuration with power off.

8. V-n DIAGRAM

Composite V-n diagrams for the three flight modes at the basic design gross weight and the minimum flying gross weight are shown in Figures VII-1 and VII-2. The diagrams for airplane flight (solid lines) were constructed as specified in MIL-A-8861 for maneuver and gust load factors.

The limit load factors for helicopter and transition flight (dashed lines) are shown as the sum of the helicopter (2.5) and the inplane load factor at a given speed, the maximums being 3.0 and -1.0.

9. LIMIT LOAD DESIGN CONDITIONS

Limit load design conditions for helicopter transition and airplane flight are contained in Table VII-2, VII-3 and VII-4, respectively. The conditions listed have been selected for investigation during preliminary design. Ground conditions to be considered are contained in Table VII-5.

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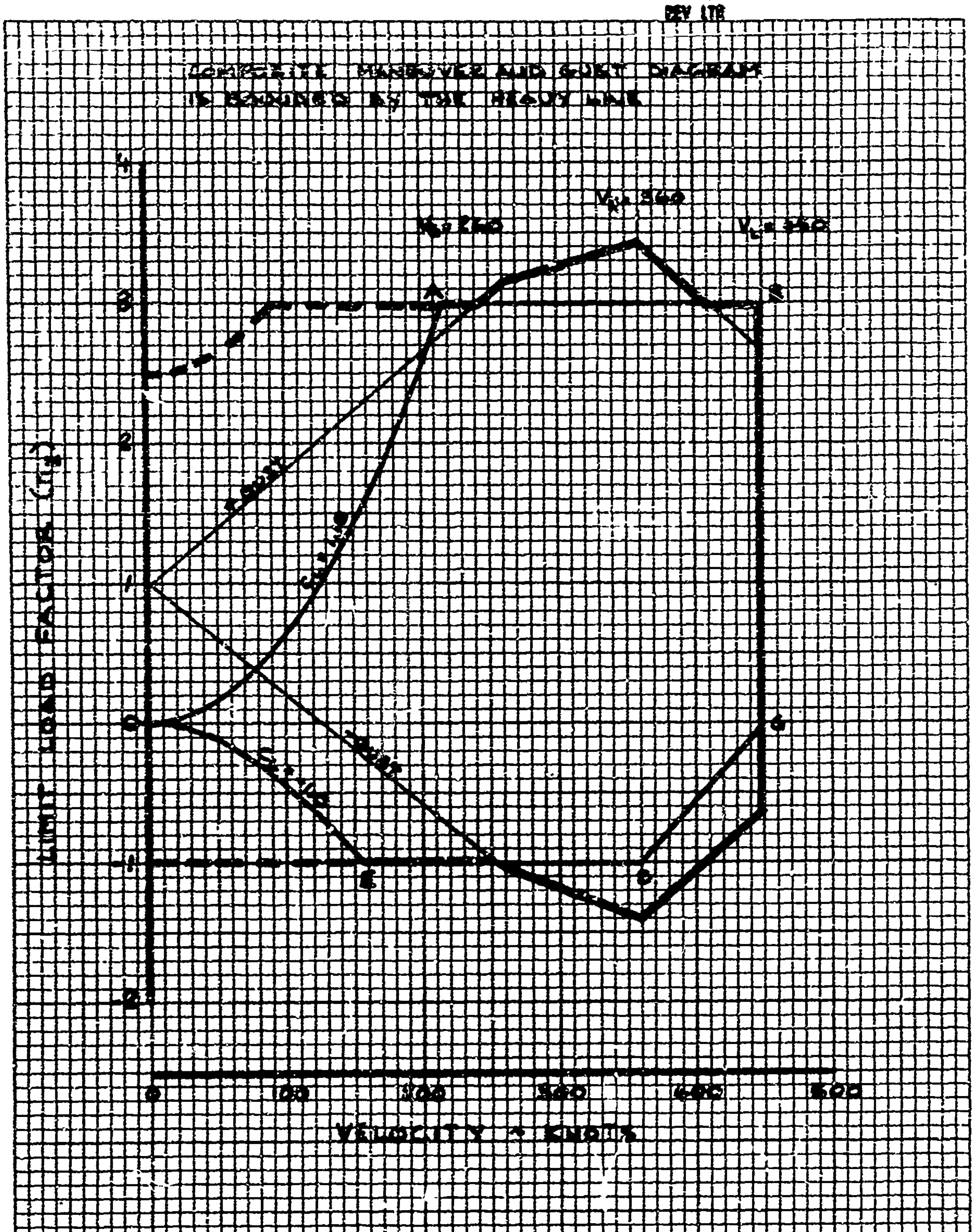


FIGURE VII-1 V - n DIAGRAM (SEA LEVEL) BASIC DESIGN GROSS WEIGHT 67,000 LB.

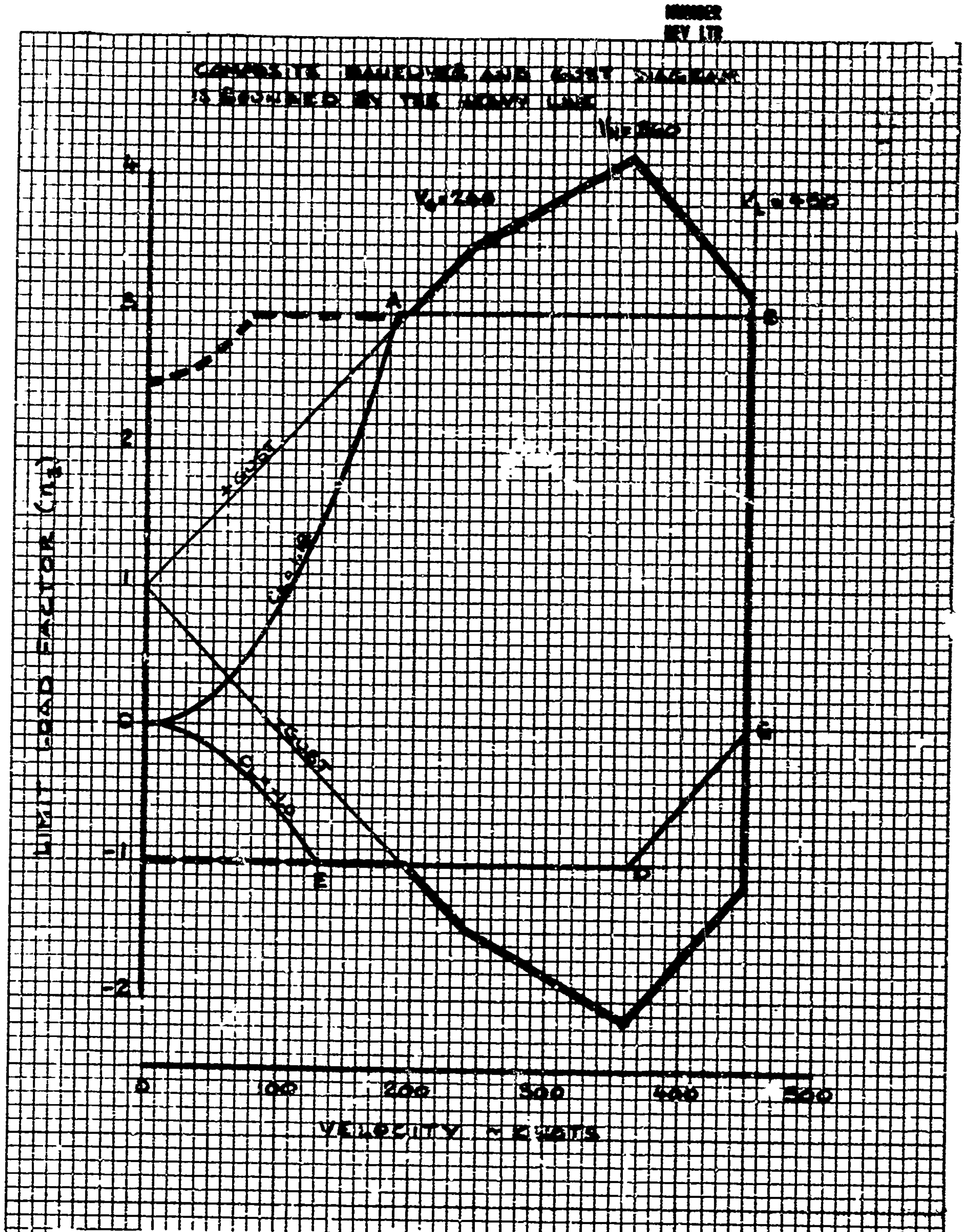


FIGURE VII-2 V - n DIAGRAM (SEA LEVEL) MINIMUM FLYING GROSS WEIGHT 47,798 LB.

TABLE VII-2 LIMIT DESIGN CONDITIONS FOR HELICOPTER FLIGHT

Cond. No.	Condition	Gross Weight Lb.	Limit Load Factor	Air Speed Knots	Resulting Rate Radians/Sec	Acceleration Radians/Sec²	Reference
1	Vertical Take-Off - NOTE 1	67,000	2.5	0	0	0	MIL-S-8698
2	Vertical Take-Off with Pitch	67,000	2.5	0	0.8 (Pitch) NOTE 2	0.6 (Pitch) NOTE 3	
3	Rolling	67,000	2.0	0	1.5 (Roll) NOTE 4	1.0 (Roll) NOTE 3	
4	Yawing	67,000	1.0	0	1.0 (Yaw) NOTE 5	.5 (Yaw) NOTE 3	
5	Pushdown (Collective Dump) - NOTE 1	67,000	-1.0	0	0	0	
6	Maximum Cyclic - NOTE 6	67,000	1.0	0	0	0	

NOTE 1: Cyclic Control shall be applied to eliminate pitching motion.
2: Maximum acceleration held until attitude is 30 degrees.
3: Maximum control input.
4: Maximum acceleration held until attitude is 60 degrees.
5: Maximum acceleration held until attitude change is 60 degrees
6: The maximum of (a) cyclic for pitch control plus half cyclic for yaw control or (b) maximum cyclic for yaw control plus half cyclic for pitch control.
7: The rotor speed for the above conditions shall be the limit rotor speed.
8: This rate results from control application.

TABLE VII-3 LIMIT DESIGN CONDITIONS FOR TRANSITION FLIGHT

Cond. No.	Condition	Gross Weight Lb.	Limit Load Factor	Airspeed Knots	Resulting Rate Rads/Sec	Acceleration Rads/Sec^2	Reference
7	Symmetrical Pull-Out	67,000	3.0	90	0.8 (Pitch)	0.6 (Pitch)	MIL-S-8698
8	Rolling Pull Out	67,000	2.4	90	1.5 (Roll)	1.0 (Roll)	MIL-S-8698
9	Yawing	67,000	1.0	90	1.0 (Yaw)	.5 (Yaw)	MIL-S-8698

NOTE 1: The rotor speed for the above conditions shall be the limit rotor speed.

2: This rate results from control application.

TABLE VII-4 LIMIT DESIGN CONDITIONS FOR AIRPLANE FLIGHT

Cond No.	Condition	Gross Weight	Limit Load Factor	Airspeed Knots	Remarks	Reference
10	Balanced Symmetrical Maneuver	67,000	[illegible].0	215	V- Diagram Point "A"	MIL-A-8861 Para. 3.2.1
			3.0	V_L		
			-1.0	155	V- Diagram Point "E"	
			-1.0	V_H		
			0	V_L		
11	Symmetrical Maneuver with Pitch	67,000	Control Displacement as Specified in MIL-A-8861, Paragraph 3.2.2.2.			
12	Rolling Pull Out	67,000	Control Displacement as Specified in MIL-A-8861, Paragraphs 3.3.1 and 3.3.1.1.			
13	Vertical Gust	67,000	As specified in MIL-A-8861, Paragraph 3.5			
		47,798				

TABLE VII-5 GROUND CONDITIONS

Cond No.	Condition	Remarks	Reference
14	Rotor Acceleration	Condition as Specified in Paragraph 3.3.1 of Referenced Spec.	MIL-S-8698
15	Landing	Landing Conditions shall be as specified in Paragraph 3.4 of referenced spec. The limit landing load factor shall be based upon a sinking speed of 12fps and rotor lift equal to two-thirds of the basic design gross weight.	MIL-S-8698

10. FATIGUE DESIGN CONDITIONS

A. Basic Fatigue Schedule

The service usage to be used for definition of preliminary design structural requirements shall be in accordance with the basic fatigue schedule Table VII-6. This schedule is based on the basic design mission. The distribution of flight time between the helicopter, transition and airplane modes is 7.9%, 12.5% and 79.6%, respectively. The total time given to maneuvers is 10.5%. The significant conditions affecting the fatigue performance of the wing are the repeated maneuvers and atmospheric turbulence at low altitudes and the relatively large number of ground-air-ground cycles. The significant conditions affecting the fatigue performance of the nacelle structure are repeated maneuvers with the vehicle in the airplane mode, ground-air-ground cycles and rotor loads. The significant conditions affecting the fatigue performance of the dynamic system are the prop/rotor cyclic control and airplane flight with inclination of the prop/rotor axis. The dynamic system of this vehicle is considered to include the prop/rotor blade, hub, controls and drive system.

TABLE VII-6

BASIC FATIGUE SCHEDULE

Condition	% Occurrence	No Per Hour
HELICOPTER MODE		
Rotor Start		1
Rotor Stop		1
Taxi	1.5	
Takeoff VTOL	.4	1
Takeoff STOL	.1	
Landing		1
Landing Flare	.5	
Hover	4.0	
Forward Flight 35 Knots	1.0	
Sideward Flight 35 Knots	.1	
Rearward Flight 35 Knots	.2	
Yawed Flight 35 Knots	.1	
Lateral Control Reversal		2
Lontigudinal Control Reversal		2
Directional Control Reversal		2
TRANSITION MODE		
Level Flight	4.0	
Climb	4.0	
Descent	2.0	
Turn 30° Bank 50 Knots	2.0	4
Pull Up 1.5G 50 Knots	.5	1
AIRPLANE MODE		
Level Flight Cruise Speed	56.0	
Level Flight Maximum Speed	4.0	
Climb	4.0	
Descent	10.0	
Maximum Power Dive at VL	.1	
Yaw at Maximum Speed	.5	
Level Turn 1.5G at VH	3.0	6
Level Turn 2.0G at VH	.5	1
Climbing Turn at VH 1.5G	.5	1
Symmetrical Pull-Up 1.5G at VH	.6	1
Symmetrical Pull-Up 2.0G at VH	.4	1

B. <u>Service Life</u>

The service life of the wing and nacelle structure shall be 10,000 hours. The service life on dynamic system components shall be 3,600 hours except as indicated below. This service life of 3,600 hours applies also to the pitch bearing. The method of analysis for the above bearings accounts for the oscillatory motion of pure rotational motion. The service life will be calculated using the cumulative damage method in conjunction with S-N curves and load/stress frequency. S-N curves for dynamic system components will be based on a mean -3σ analysis. S-N curves for the wing structure shall be established using the mean of data associated with the most critical stress concentration. The calculated "mean life" thus established will be divided by a scatter factor of four (4) to establish a "safe life" on service life.

The B_{10} design life for the individual drive system bearings will be established based on the desired transmission Mean Time Between Removal (MTBR). This means that the total bearing system life, when combined with other critical component lives will result in the desired transmission MTBR.

Gear box cases shall be designed for a service life of 10,000 hours considering drive train and rotor loads. All drive system gearing and splines shall be designed for unrestricted fatigue life under maximum rated power at normal operating RPM.

C. Take-Off Condition

A vertical load take-off spectrum shall be used for the take-off phases of the fatigue schedule.

D. Landing Condition

A spectrum of landing sinking speeds shall be used for the landing phase of the fatigue schedule.

E. Taxi Condition

A vertical load taxi spectrum shall be used for the taxi phases of the fatigue schedule.

F. Gust Condition

A gust load spectrum shall be used as specified in MIL-A-8866(ASG), Paragraph 3.4

11. WING DESIGN CRITERIA

A. Flight and Ground Loads

The wing shall be designed for the various flight and ground load conditions defined in paragraphs 9. Dynamic analyses shall be used to obtain the wing loads due to gusts paragraphs 10-F and the landings paragraph 10-D.

B. Rotor System Loads

The wing shall be designed for the loads due to the rotor system as defined in paragraph 13.

C. Fatigue Considerations

The primary structure of the wing shall be analyzed to determine its fatigue performance under the conditions specified in paragraph 10 , which includes loadings caused by gusts, maneuvers, landing, take-off and taxing. The wing shall incorporate fail-safe design.

12. NACELLE STRUCTURE DESIGN CRITERIA

A. Flight and Ground Loads

The nacelle structure, which includes the tilt mechanism and wing attachment, shall be designed for the various flight and ground loads defined in Paragraph 9. In addition, the nacelle structure shall be designed for a side load factor equal to $\pm$ 2.0 limit with the nacelle in the vertical and horizontal positions.

B. Rotor System Loads

The nacelle structure shall be for the rotor system loads as defined in paragraph 13.

C. Drive System Loads

The nacelle structure shall be designed for the limit torque load condition as specified in paragraph 14.

D. Fatigue Considerations

The nacelle structure shall be analyzed to determine its fatigue performance under the conditions specified in paragraph 10 which includes gusts, maneuvers, landing, take-off, taxiing and rotor vibratory loads. In addition, loads due to aircraft rates and accelerations will be considered. The nacelle structure shall incorporate fail safe design.

13. ROTOR SYSTEM DESIGN CRITERIA

A. Flight Loads

The prop/rotor blade, hub and controls shall be designed for the various flight conditions defined in paragraph 9. The loads including vibratory and steady shall be calculated by aeroelastic analysis.

B. <u>Fatigue Considerations</u>

The rotor system shall be analyzed to determine its fatigue performance under the conditions specified in paragraph 10. In addition, the following criteria will be used:

a. Alternating loads due to rotor cyclic control. in the helicopter mode, equal to the cyclic required to trim the aircraft level plus 25% of the maximum cyclic for pitch control shall not exceed the fatigue endurance limits of rotor system components.

b. Alternating loads due to rotor cyclic control, in the helicopter mode, equal to the cyclic required to trim the aircraft level plus 25% of the maximum cyclic for yaw control shall not exceed the fatigue endurance limits of rotor system components.

c. Alternating loads due to "Aq" equal to 1,500 shall not exceed the fatigue endurance limits of rotor system components.

14. DRIVE SYSTEM DESIGN CRITERIA

A. <u>Limit Design Loads</u>

The drive system includes all components of the drive train between and including the engine shafts and the main rotor shafts with all engines at maximum rated power and at normal operating RPM. The torque split between rotors shall be 75-25 in combination with rotor loads defined in paragraph 13.

B. Fatigue Considerations

Bearing B_{10} life shall be based on cubic mean loads for the conditions of the Basic Fatigue Schedule Table VII-6.

All gears and spurs within the drive train shall be designed for unrestricted fatigue life under maximum transmission ratio power or maximum rated engine power, whichever is lower, at normal operating RPM. The alternating torque for symmetrical flight conditions shall be considered to be ± 15% of the steady torque. This alternating torque shall be considered for the design of transmission and shafting exclusive of the gear teeth and bearings.

15. MATERIALS AND ALLOWABLES

A. Material Selection

The increased knowledge of strength and reliability of new materials and construction techniques contributing to a general advance in the "state of the art" shall be incorporated in the design wherever feasible. Materials shall be selected on the basis of technical suitability to satisfy design requirements of function, reliability, strength safety, fabrication ease and economy. Particular attention shall be given to material work propagation, fracture toughness and corrosion characteristic and to protective finish systems and processes for the prevention and control of corrosion and stress corrosion. Materials to be considered, but not limited to, include:

a. for wing fatigue critical areas titanium **some heats of** 6AL-4V and/or 2024 aluminum alloy for skin and stringer combinations and titanium 6Al-4V and/or 7079, 7175 and alloy 71 aluminum alloys for forgings and thick machined plates. Figure VII-3 shows that improved stress corrosion characteristics of high strength 7175 and alloy 71 aluminum alloys over the alloys in current use.

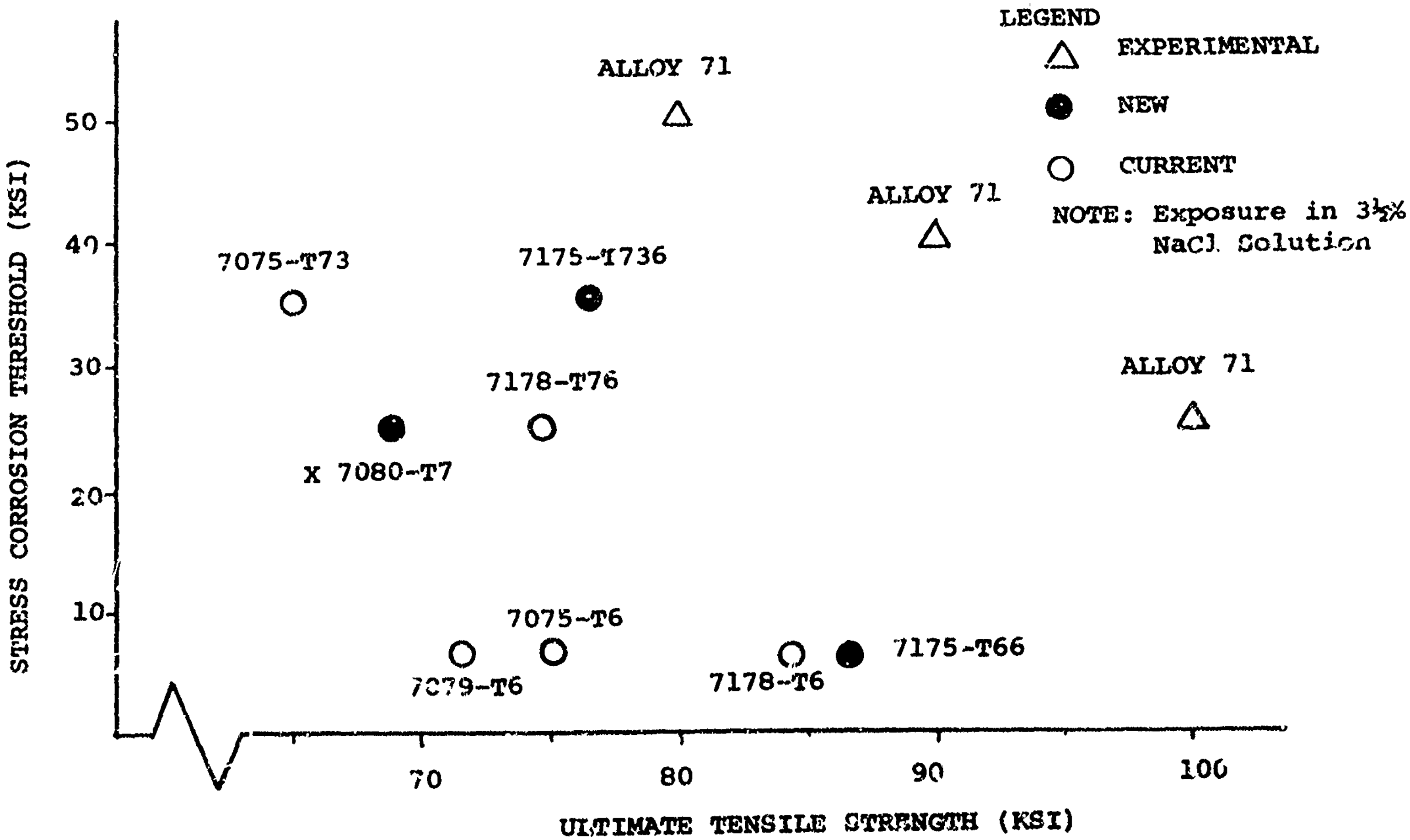


FIGURE VII-3. STRESS CORROSION THRESHOLD FOR ALUMINUM FORGINGS

b. for wing non-fatigue critical areas some heats of titanium 6AL-4V and/ or 7075, 7178, 7175 aluminum alloys.

c. for the rotor/prop blade a composite structure consisting of a fiberglass spar, cross-ply skins, aluminum honey-comb core and a titanium leading edge erosion strip. Fatigue properties of S-glass composites based on testing conducted at Boeing-Vertol are shown in Table VII-7 and Figures VII-4 and -5.

d. for the rotor hub 6Al-4V titanium forging.

e. for the transmission gears VASCO X2 steel.

f. for transmission bearings M50 vacuum melt steel, 52100 vacuum melt steel and carburized steels.

g. for the transmission case magnesium, steel, titanium, composites and metal matrix composites. Figure VII-6 shows the improved fatigue properties of rare earth east magnesium alloy ZE63-T6 over those for magnesium alloys in present use.

TABLE VII-7
A SUMMARY OF THE MECHANICAL AND PHYSICAL PROPERTIES OF "S" GLASS-FIBER EPOXY-RESIN LAMINATES

MATERIAL DESCRIPTION	FILAMENT ORIENTATION WITH RESPECT TO LOAD AXIS	THICKNESS PER PLY (INCHES)	ALLOWABLE TENSILE STRESS (KSI)	INITIAL ELASTIC MODULUS X 10^6 PSI	ALLOWABLE ALTERNATING FATIGUE STRESS FOR R=0.1		
					AT 10^6 CYCLES (KSI)	AT 10^7 CYCLES (KSI)	AT 10^8 CYCLES (KSI)
1002S	0° (1)	.0093	175	(4) 6.30	11.1	9.05	7.36
1002S	±45° (2)	.0099	28.2	1.59	2.21	2.06	1.92
1002S	90° (3)	.0112	3.41	1.78	.500	.431	.371
XP251S	0°	.0078	239	(4) 7.45	14.1	11.9	9.96
XP251S	±45°	.0080	22.6	(4) 2.40	1.89	1.64	1.43
BP907-143S	0° (5)	.0102	150	5.08	5.51	4.12	3.08
BP907-143S	±45° (5)	.0106	30.6	2.10	2.47	2.29	2.13

(1) Filament Orientation 0°, 0°, 0°, 0°, 0°, 0°, 0°

(2) Filament Orientation +45°, -45°, +45°, -45°, +45°, -45°

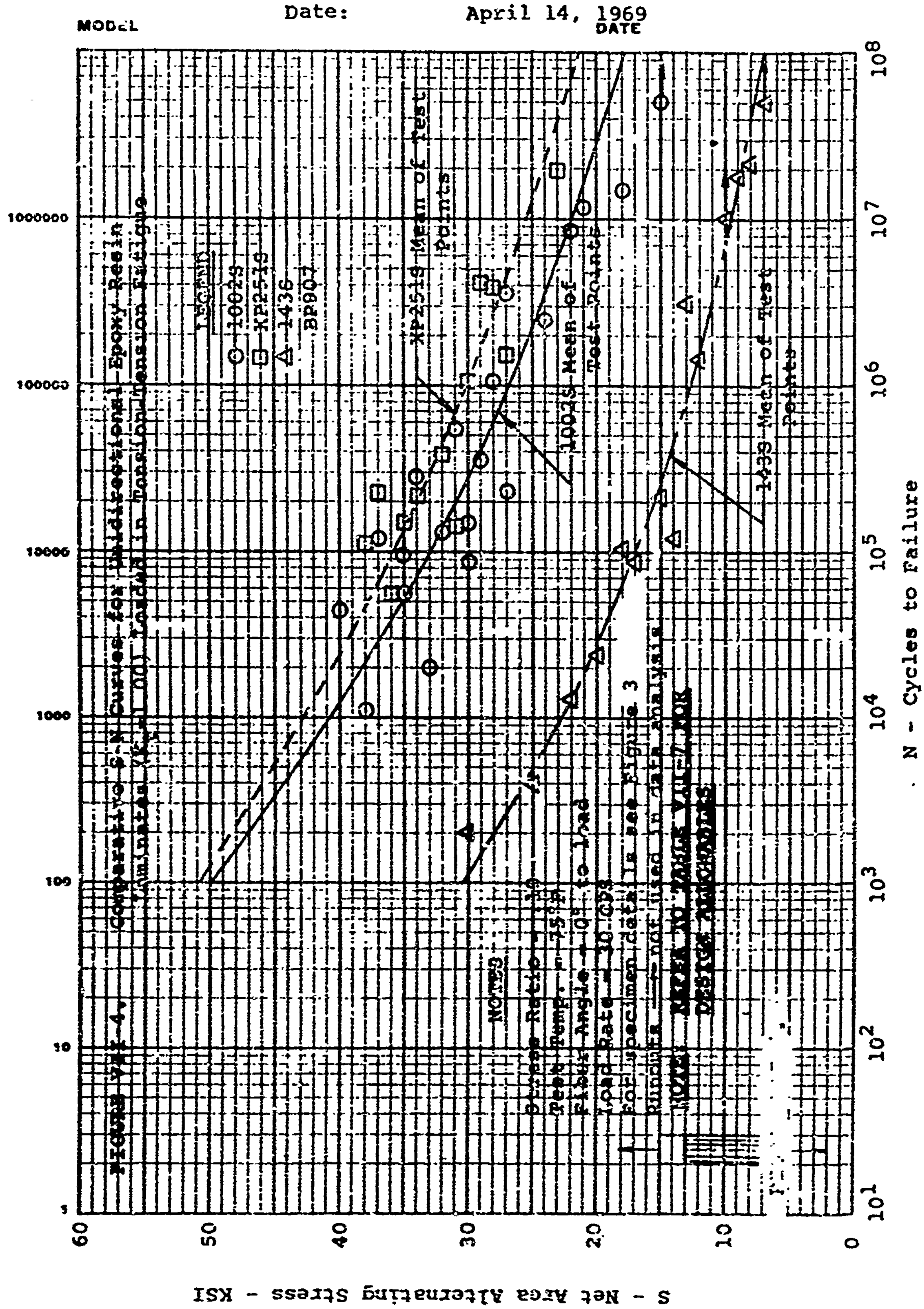
(3) Filament Orientation 90°, 90°, 90°, 90°, 90°, 90°

(4) These elastic moduli are adjusted so that they comply with those values obtained from full scale component testing.

(5) Refers to warp direction only.

FORM 11180 (0/67)

Prepared by: T. Patterson
Checked by: R. Jacobs, S. Beshore
Approved by: R. White
Date: April 14, 1969



Prepared by: T. Patterson
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Approved by: R. White
Date: April 14, 1969

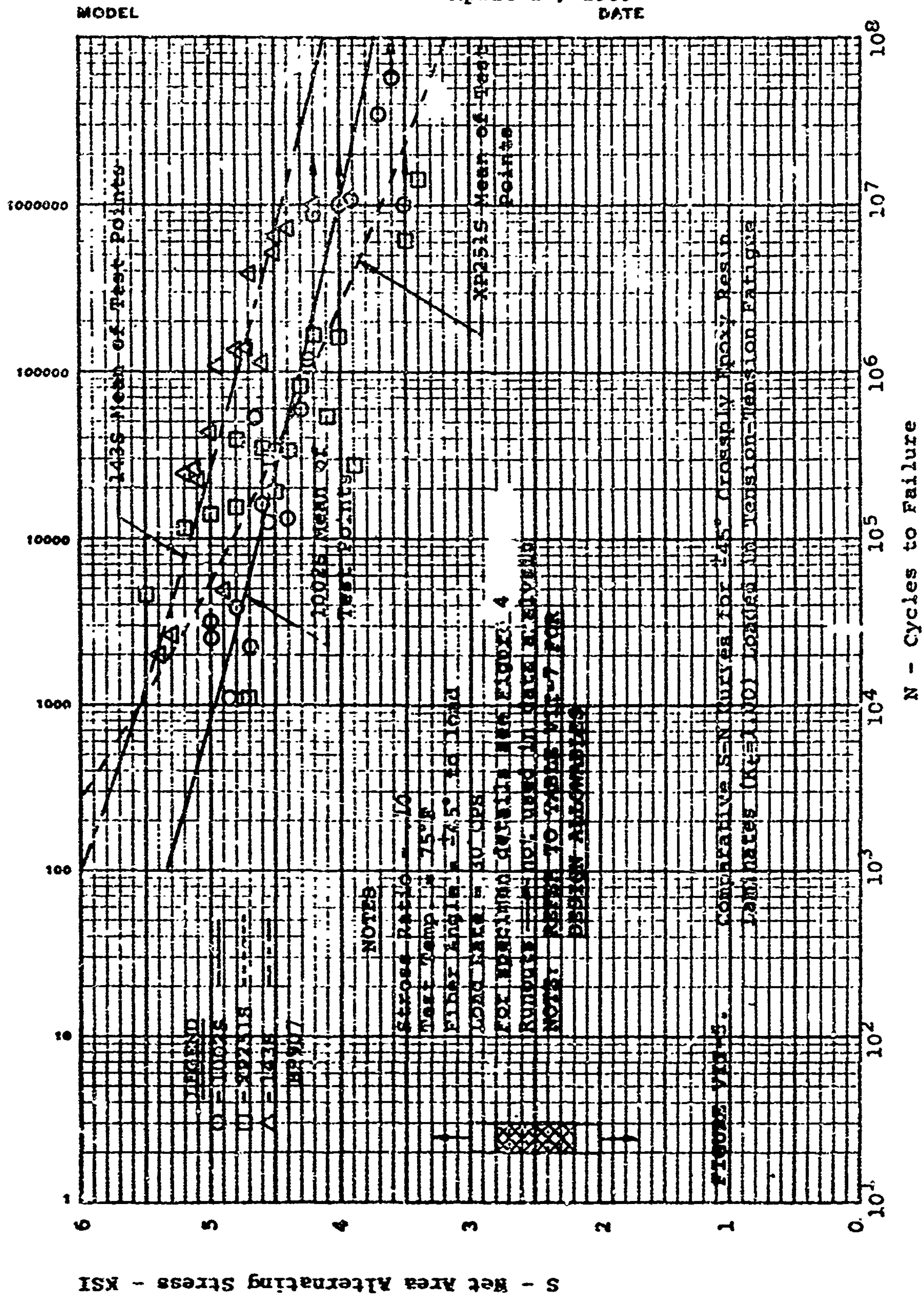


FIGURE VII-5. Comparative S-N Curves for ±45° Crossply Epoxy Resin Laminates (Kt=1.0) Loaded in Tension-Tension Fatigue

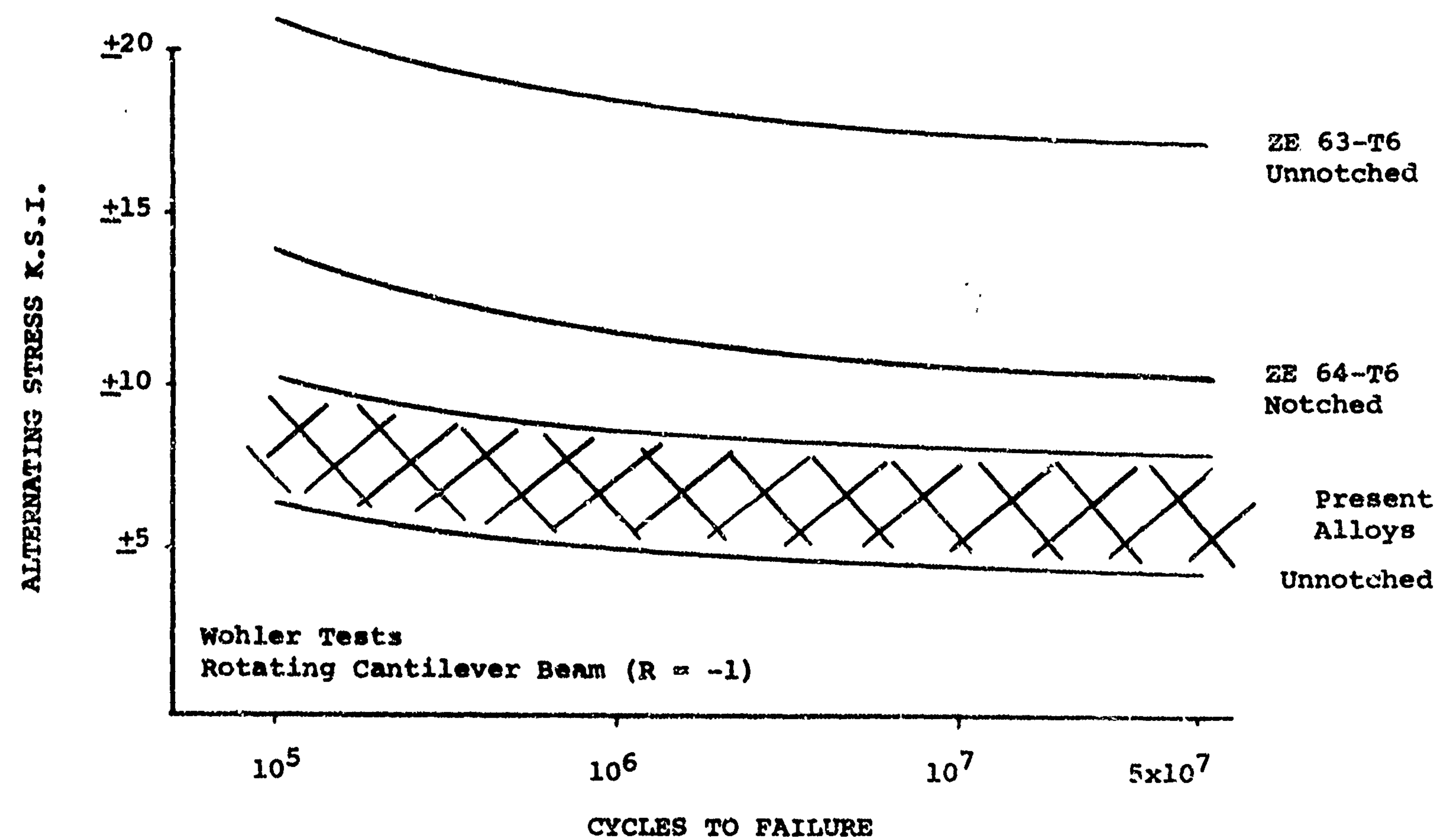


FIGURE VII-6. FATIGUE STENGTH OF CAST MAGNESIUM ALLOYS

B. Material Allowables

Material strength properties will be based upon the following:

a. Anticipated design allowables for new materials based on preliminary data consistent with 1972 technology.

b. MIL-HDBK-5 Metallic Materials and Elements for Flight Vehicles. Column "B" allowable stresses will be used where failure of an individual element would result in the applied load being safely distributed to other load carrying members. In all other applications, the Column "A" values will be used.

c. MIL-HDBK-17, "Plastics for Flight Vehicles".

d. MIL-HDBK-23, "Composite Construction for Flight Vehicles".

e. Boeing-Vertol Structure Design Manual.

f. Boeing-Vertol Report SRR-7, "Reinforced Composite Material Allowables". This document contains design strength and mechanical properties used at Boeing-Vertol for Boron and S-Glass composites.

16. AEROELASTIC STABILITY

An analysis has been made to ensure that there are no whirl flutter or air/ground resonance problems with the USAF Tilt Rotor aircraft. Whirl flutter and air/ground resonance prevention have been treated in this study since wing and/or nacelle stiffnesses could significantly increase the weight. Since the configuration analyzed is adequately stable, the results of the trend weights used in the performance studies are believed to be valid. This result is due in part to the provision of cyclic feedback in the rotor control systems (the LARMS system described in Section VI,3).

Rotor blade aeroelastic stability has not been treated in this study except for the consideration of stall flutter made in Section IV,5. Blade design is to be pursued in detail in Phase II and will be treated at that time. Experience with other designs using the soft-inplane hingeless blade approach has shown this design to be practicable and has substantiated the rotor weight trends used in this study. The parameters of the aircraft analyzed are summarized for reference purposes in Table VII-8.

TABLE VII-8

PARAMETERS OF AIRCRAFT USED FOR AEROELASTIC STABILITY ANALYSIS

DESCRIPTION	UNITS	VALUE
Radius of Rotor	Inches	330
Number of Blades	N.D.	3
First Moment of 1 Blade About Flap Hinge	Lb-Sec^2	85.55
Inertia of 1 Blade about Flap Hinge	Lb-Sec^{2}In.	13,842
Ratio of Blade Cut Out to R	N.D.	0.2
Blade Twist at 75%R (Root Reference)	Deg.	-16.
Mean Chord	Inches	32.3
Lift Slope Coefficient	1/Rad	5.73
Distance from Center of Hub to Nacelle Pivot	Inches	112.
Distance Between Nacelle Pivot and Effective Wing Root (Approx. to be 61% of Wing Semi Span)	Inches	212
Distance Between Nacelle Pivot and cg of Rotor Nacelle Combination	Inches	67.4
Nacelle (Including Blades & Hub) Moment of Inertia in Pitch	Lb-Sec^2-In.	164683
Weight of Nacelle Including 3 Blades and Hub	Lb	9500
Wing/Nacelle Pitch (Torsion) Frequency	Cps	2.75
Wing/Nacelle Yaw (Chordwise) Frequency	Cps	4.36
Wing/Nacelle Vertical Bending Frequency	Cps	1.68
Rotor Speed - Cruise	Rpm	183.
Forward Speed - Aircraft Cruise	Knots	350

(Continued on Following Page)

DESCRIPTION	UNITS	VALUE
Lateral Stiffnesses of Rear Tires (same)	Lb/Ft	119,000
Lateral Stiffness of Front Tire	Lb/Ft	98,000
Vertical Stiffness Rear Tires	Lb/Ft	83,000
Vertical Stiffness Front Tire	Lb/Ft	68,200
Fwd/Aft Stiffness Rear Tires	Lb/Ft	324,000
Fwd/Aft Stiffness Front Tire	Lb/Ft	144,000
Landing Gear Damping in Vertical Direction All Tires Same	Lb-Sec./Ft	135.
Blade Flap Frequency	Cps	4.09
Blade α of attack at 75% Radius	Deg.	0
Effective Hinge Offset	In.	66

NOTES:

1. Blade parameters used were for the TRB-3B Design.
2. The six degree of freedom analysis computer program (C-26) was used for the whirl flutter analysis
3. Computer program C-2[illegible] was used for the ground resonance analysis

A. <u>Ground Resonance Stability</u>

The tilt rotor aircraft with soft-inplane hingeless rotors can have ground resonance stability problems due to blade chordwise (lag) bending coupling with an airframe or landing gear mode. Such resonance conditions must be damped by the landing gear oleos, airframe and blade structural damping and rotor blade aerodynamic damping. As shown in Figure VII-7, there are two regions of instability possible if this damping was zero but if nominal, values of damping are assumed the aircraft is stable.

The upper graph of Figure VII-7 shows four regions of coalescence of rotor and aircraft frequencies as a function of rotor speed. Instabilities might be expected at any of these intersections. In fact, considering zero blade and structural damping which is conservative, the only unstable situations occur at the lower rotor-wing vertical bending frequency intersection (near hover rpm) and at the lower rotor-wing chordwise bending frequency intersection. For nominal damping (2% structural damping and rotor aerodynamic damping effects considered) these instabilities

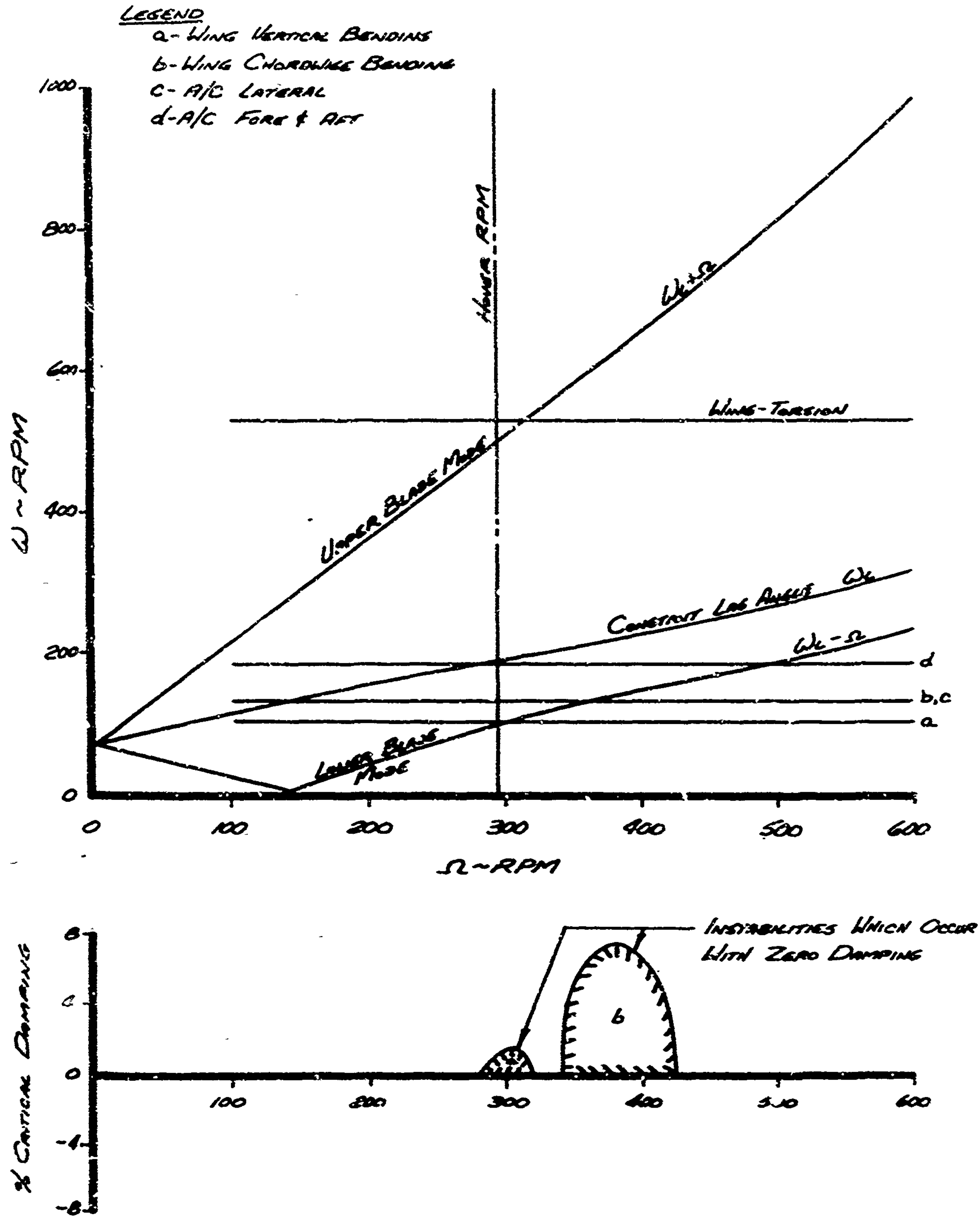


FIGURE VII-7 GROUND RESONANCE MODES ARE STABLE IN OPERATING RPM RANGE WITH NOMINAL DAMPING are eliminated. Previous studies on similar configurations have shown a/c lateral and a/c fore and aft motions to be the only rigid body modes tending to produce ground resonance. For this aircraft configuration these modes are stable. The analysis used for this study is described as follows:

Program C-27: A multi-bladed rotor is considered with motion of the blades described by two arbitrary blade mode shapes having components parallel and perpendicular to the blade root chord. The dynamic system (Figure VII-8) has nine degrees of freedom. These freedoms are:

- P = Nacelle Pitch and Wing Torsion
- Y = Nacelle Yaw and Wing Chordwise Bending
- R = Nacelle Roll and Wing Flapwise Bending
- F_{10} = Constant out of plane blade bending tip deflection of first mode (related to coning angle)
- F_{IC} = Pitch of Tip Path Plane of Mode One
- F_{IS} = Yaw of Tip Path Plane of Mode One
- F_{20} = Constant out of plane bending tip deflection of second mode (related to coning angle)
- F_{2C} = Pitch of Tip Path Plane of Mode Two

F_{2S} = Yaw of Tip Path Plane of Mode Two

The nine Lagrangian equations of motion were expressed in matrix form and linearized by an expansion in a Taylor series about the equilibrium point ($\ddot{q} = \dot{q} = q = 0$) and a retaining of only the first order terms. The program assumes a linear rotor blade lift equation, zero rotor blade drag and zero wing aerodynamics.

Blade arbitrary mode deflections can be defined for up to ten sections. The program has the provision that collective pitch can be calculated such that the blade angle of attach at .75 blade radius can be specified.

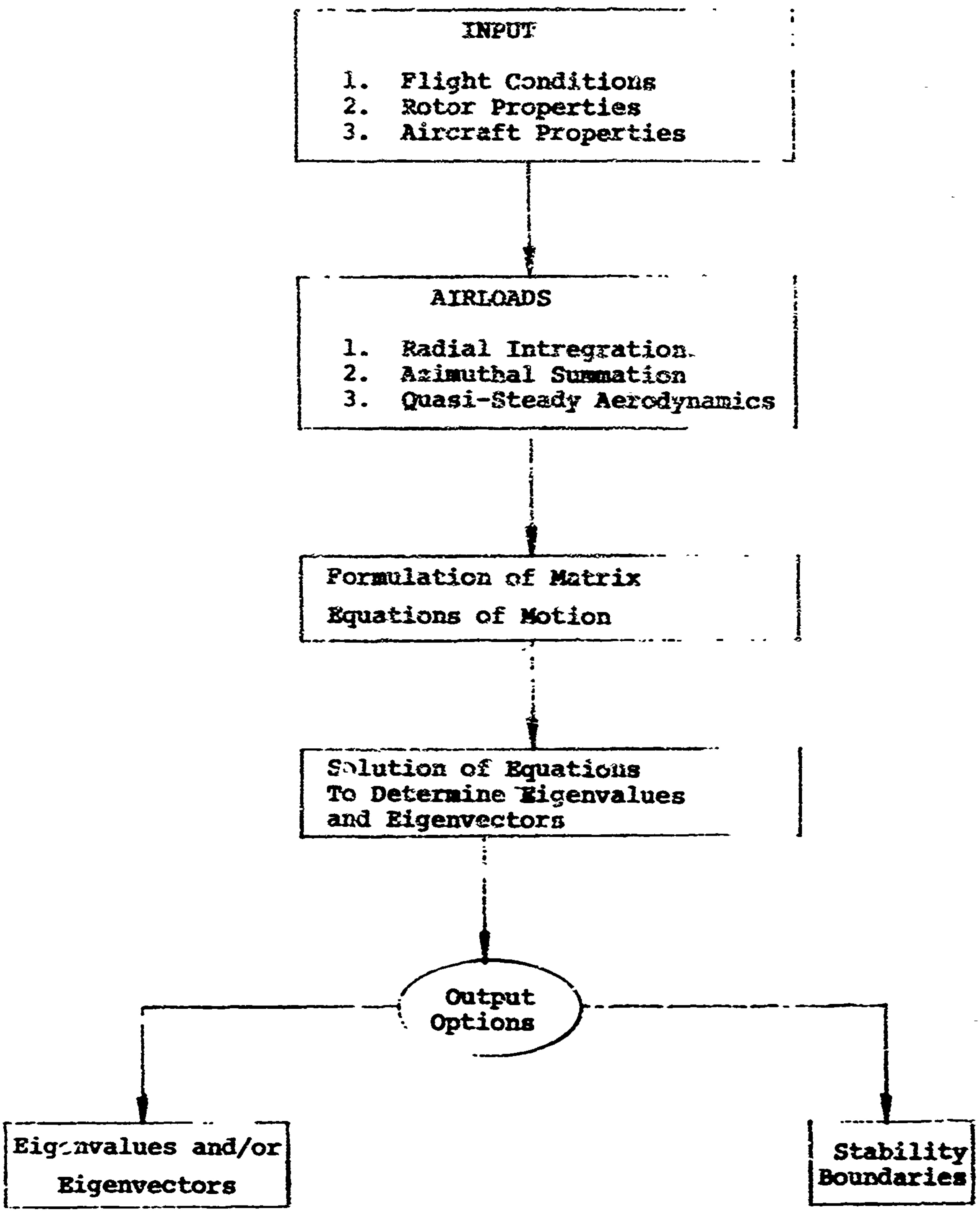
Print-out options include stability boundaries, eigenvalues, and eigenvectors for variations of parameters such as inflow ratio, pitch frequency, yaw frequency, roll frequency, etc., in a single run.

B. Whirl Flutter

Results of a study with wing/nacelle yaw frequency and wing/nacelle pitch frequency varying and other parameters fixed at nominal are shown by Figure VII-9. The Model 215 aircraft was considered to be in the nominal cruise flight mode, 350 kt. (EAS), with no control feedbacks. The aircraft design is stable.

As can be seen by Figure VII-9, a very significant parameter for both whirl flutter and divergence is the wing torsional stiffness and corresponding frequency. For nominal aircraft properties, increasing the wing/nacelle torsional stiffness significantly improves the stability of the system. The wing/nacelle chordwise bending stiffness has a relatively minor effect on the stability boundaries for practical variations around nominal.

C-27 STABILITY ANALYSIS



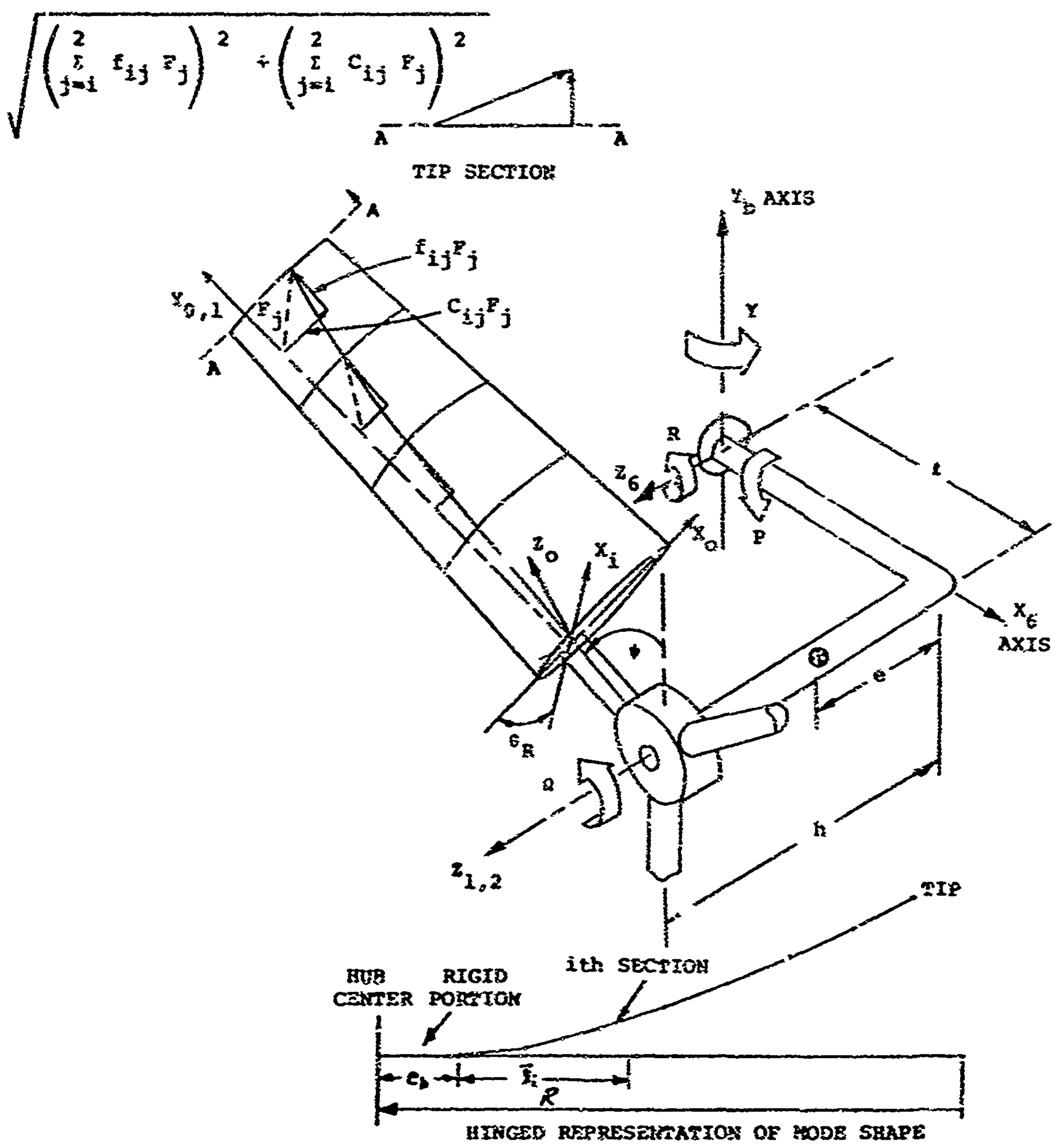


FIGURE VII-8 NINE DEGREE-OF-FREEDOM PROPELLER WHIRL MODEL ANALYSIS

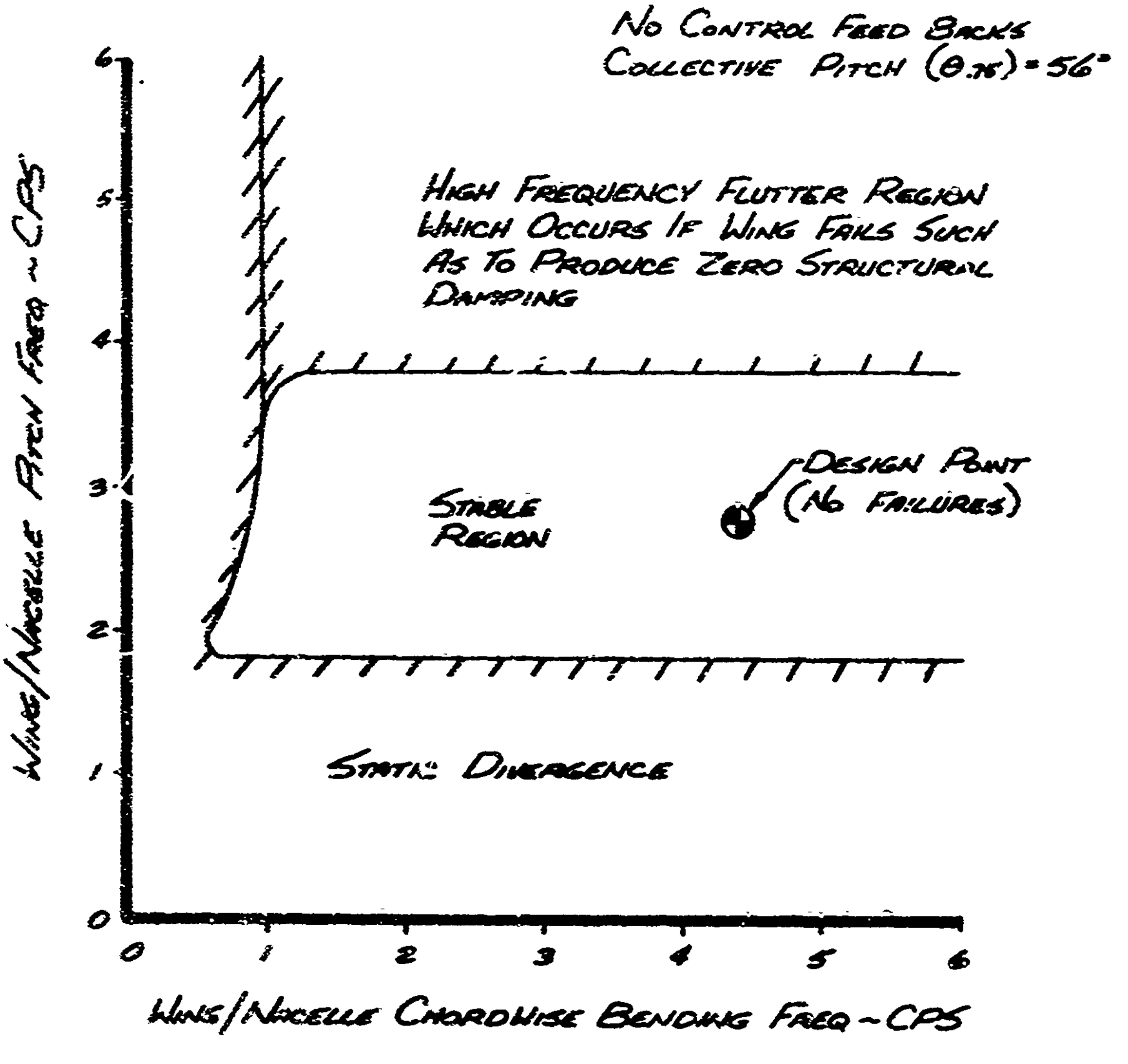


FIGURE VII-9 MODEL 215 DESIGN IS STABLE FROM WHIRL FLUTTER AT 350 KNOTS (EAS) SPEED WITH CYCLIC FEEDBACK SYSTEM INOPERATIVE

The high frequency flutter region shown is present only if the structural damping is assumed zero. This flutter is high frequency (greater than 2 cps) forward whirl flutter. For normal (2 per cent) structural damping in the wing/nacelle vertical bending, wing/nacelle chordwise bending, and the wing nacelle torsion mode this whirl flutter region becomes stable. This implies that this flutter does not exist under normal conditions of structural damping even without cyclic feedback. Wind tunnel tests on models of similar configurations have verified this, as high frequency flutter was not encountered.

The rotor speed margin of the aircraft is adequate at the cruise velocity of 350 kt (EAS). As shown in Figure VII-10 this margin is approximately 45 rpm. The aircraft stability is quite sensitive to rotor rpm above 200 rpm if the wing/nacelle pitch frequency was reduced. The flutter region shown is slightly negative damped but is avoided by a good margin with the present design.

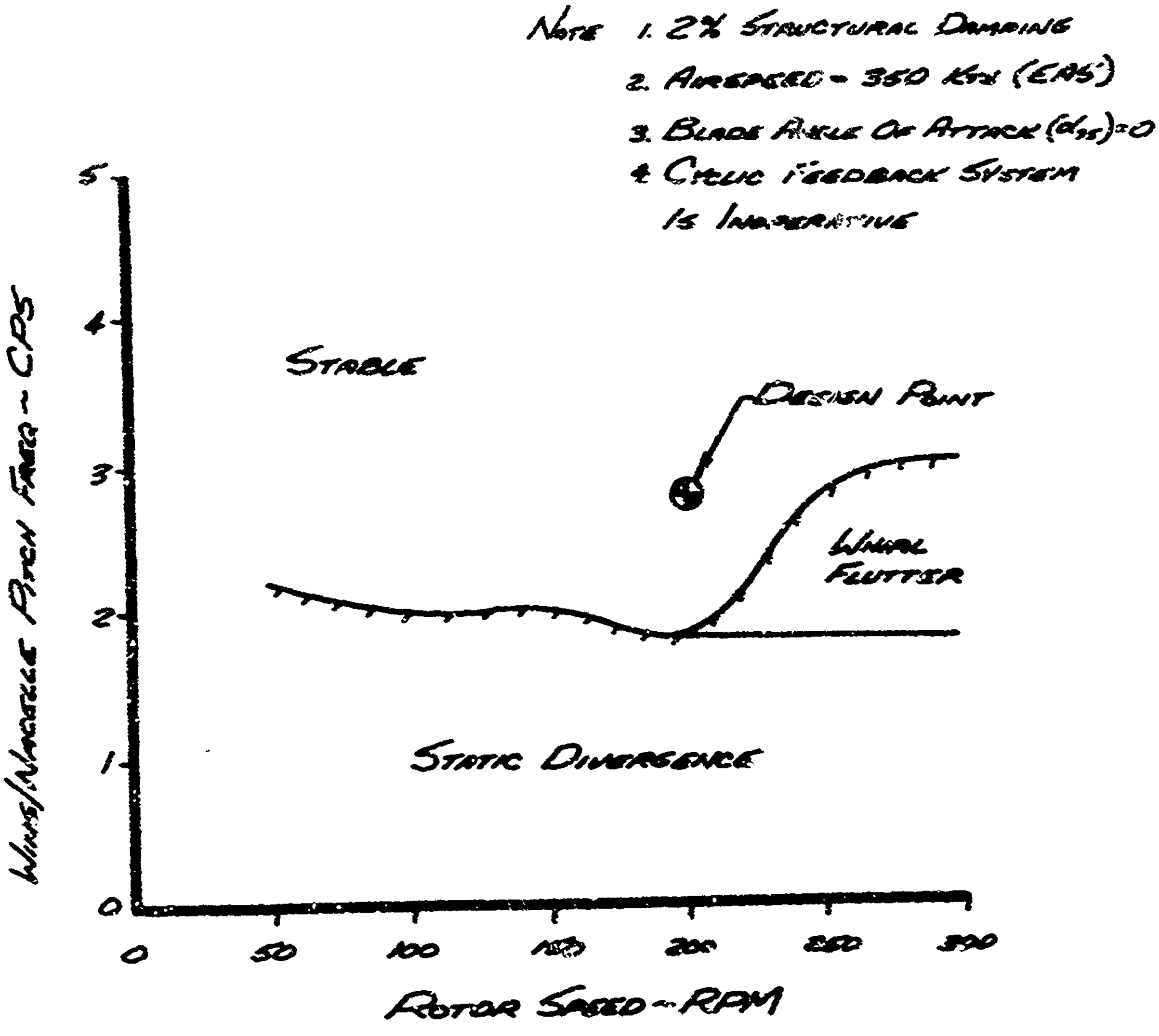


FIGURE VII-10 ROTOR SPEED MARGIN OF AIRCRAFT DESIGN IS ADEQUATE AT 350 KTS. (EAS)

Dash speed capability reduces the static divergence stability velocity margin as shown in Figure VII-11. For a dash speed of 400 kt. (EAS), the margin is approximately 100 kt. This figure also emphasizes again the importance of wing/nacelle pitch stiffness (or frequency) on whirl flutter/divergence safety margins.

A low power setting at near dash speeds can produce a static divergence problem requiring the cyclic pitch feedback as shown in Figure VII-12. The propellers could approach a wind illing condition during slowdown from dash speed and can produce an unsafe condition if the cyclic system were not provided.

The analytical model used for this study is shown in Figure VII-13. This is a 6-degree-of-freedom analysis which describes the blade coning, pitch and yaw of the disc plane, wing/nacelle vertical bending (vertical translation), torsion (wing/nacelle pitch), and chordwise bending (wing-nacelle yaw). The capability of treating both the effects of structural damping and feathering feedback are included. The analysis computes the stability boundary as a function of

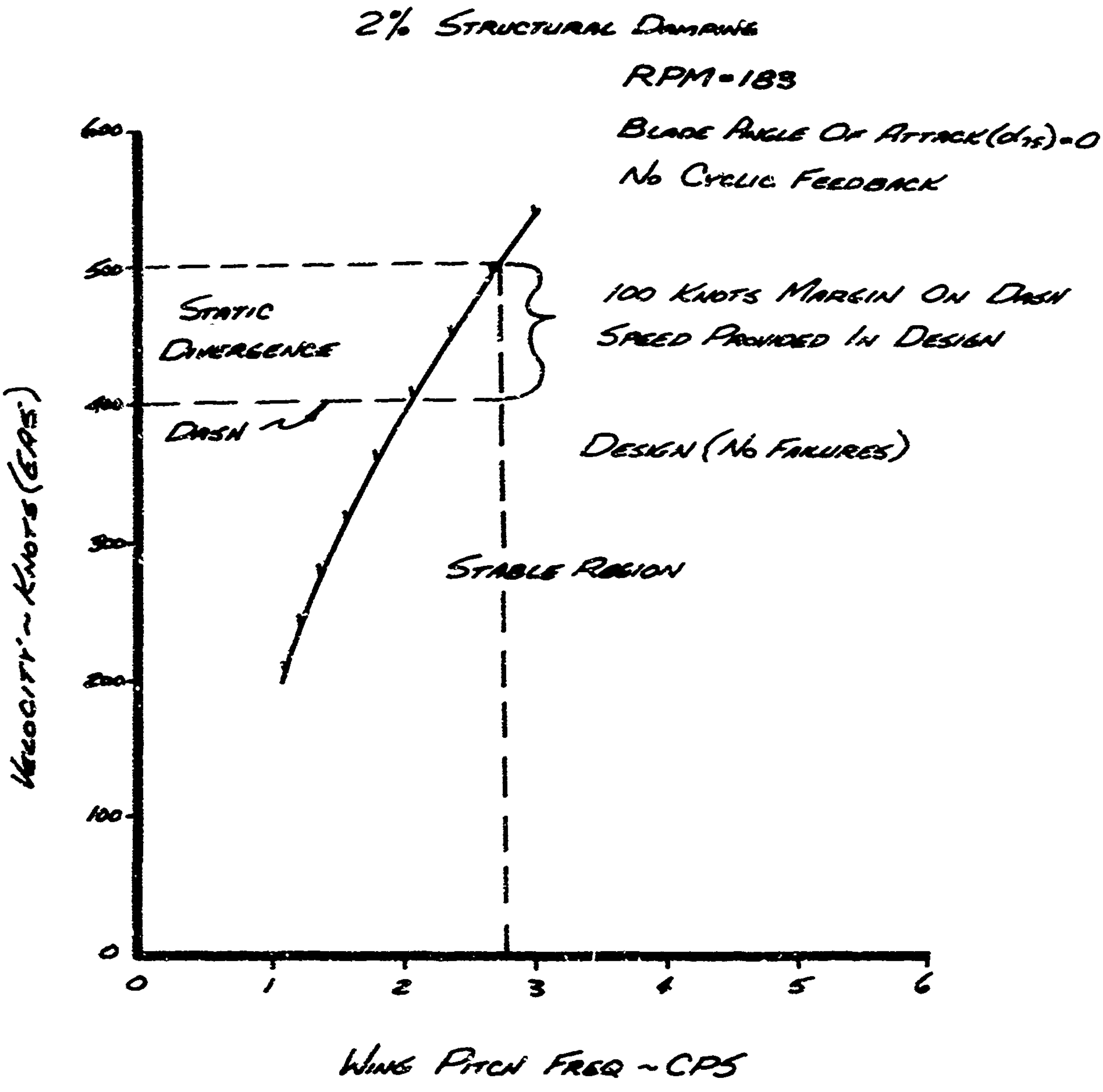


FIGURE VII-11 DASH SPEED CAPABILITY REDUCES STABILITY MARGIN

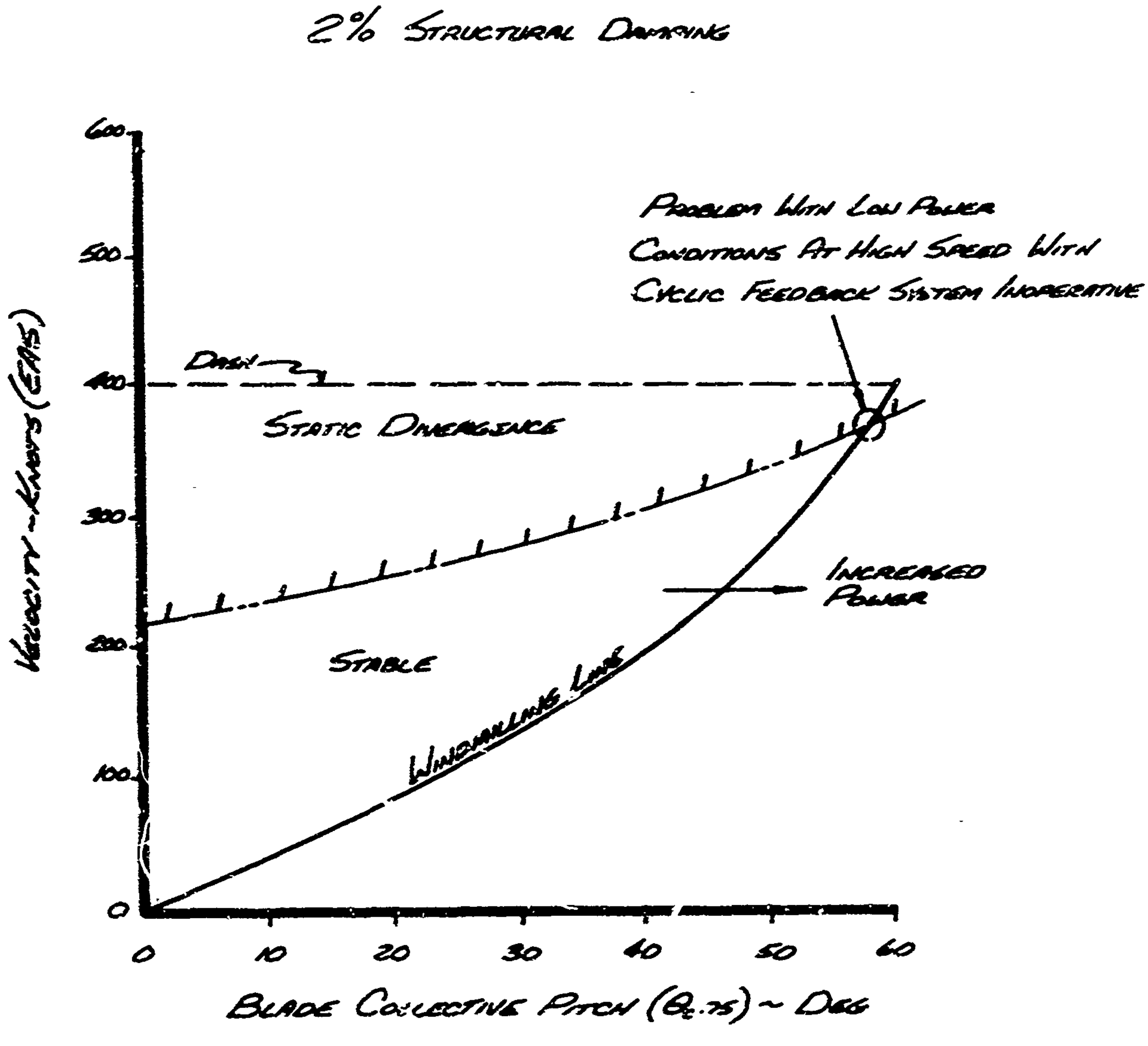
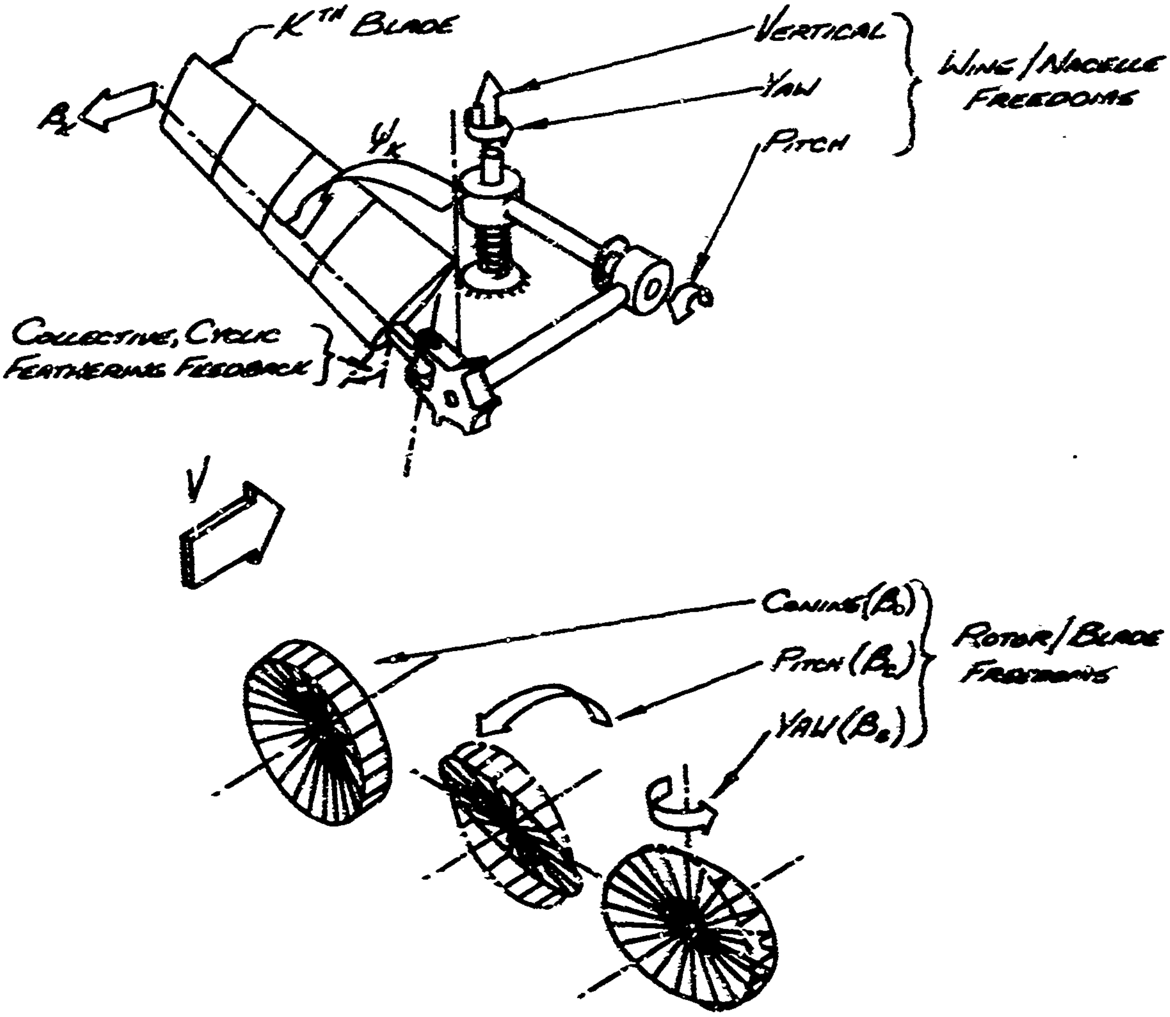


FIGURE VII-12 LOW POWER SETTING AT DASH SPEED PRODUCES STATIC DIVERGENCE PROBLEM REQUIRING CYCLIC PITCH FEEDBACK



MOTION OF THE K^{TH} BLADE:

$$\beta_k = \beta_0 + \beta_c \cos\left\{\psi_k + \frac{2\pi}{n}(K-1)\right\} + \beta_s \sin\left\{\psi_k + \frac{2\pi}{n}(K-1)\right\}$$

FIGURE VII-13 ANALYTICAL MODEL USED FOR THIS ANALYSIS variation in pitch and yaw natural frequencies.

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13. ABSTRACT - Basic design studies on prop/rotor aircraft performed as the first phase of the four phase USAF Contract F33615-69-C-1570 are summarized in this interim report. This program is to determine design criteria and demonstrate the adequacy of technology by designing a full-scale prop/rotor aircraft and by designing, manufacturing and testing scaled models. Selected from preliminary design and performance sensitivity tradeoffs was a prop/rotor aircraft for a transport mission with a 250 N. Mile radius, a cruise speed of 350 knots and a payload of 5 tons with a VTO at 2500 ft and 93°F. This aircraft can also perform a rescue mission with a 500 N. Mile radius and a mid-point hover time of 30 minutes. The Landing Gear was sized for a coverage of 40 and 38 passes when operated on CBR4 soil. A 21% wing thickness is used to provide a wing compatible with high speed drag rise and to satisfy the structural requirements with a minimum weight wing. The prop/rotor utilized has no flap or lag hinges. Rotor blade cyclic pitch is planned to provide both control moments and load alleviation. A hover figure of merit of 75% and a cruise efficiency of 78% are expected for this aircraft. A useful load fraction of 31.6% is projected, based on conservative estimates.

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14. KEY WORDS	LINK A		LINK B		LINK C	
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