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USAAVLABS TECHNICAL REPORT 65-71

EXTENSION OF THE HIGH-SPEED FLIGHT ENVELOPE OF THE XH-51A COMPOUND HELICOPTER

By

Donald R. Wyrick

November 1965

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-150(T)
LOCKHEED-CALIFORNIA COMPANY

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The Army is currently sponsoring other high-speed rotary-wing flight research programs similar to this to provide the basic technology required for future designs of high performance rotary-wing aircraft.

Task 1P121401A14311

Contract DA 44-177-AMC-150(T)

USAAVLABS Technical Report 65-71

November 1965

EXTENSION OF THE HIGH-SPEED FLIGHT ENVELOPE

OF THE XH-51A COMPOUND HELICOPTER

L.R. 18917

by

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ABSTRACT

This report presents the results of a speed extension program conducted by the Lockheed-California Company on the compound version of the rigid-rotor XH-51A helicopter (under a supplement to contract DA 44-177-AMC-150(T), Modification 1). The principle objective of this program was to investigate the flight characteristics of the compound helicopter with special emphasis on the areas of flying qualities, performance, structural loads, vibration, and maneuverability in the speed range of 200 to 230 knots true airspeed. All contract objectives were met or exceeded. The maximum level flight true airspeeds demonstrated, during this program, were 236 knots at sea level and 228 knots at a density altitude of 12,000 feet.

FOREWORD

This report describes a program of research in high-speed flight with the XH-51A Compound Rigid Rotor Helicopter. The research program was conducted by the Lockheed-California Company under contract to the U. S. Army Aviation Materiel Laboratories (USAAVLABS), Fort Eustis, Virginia.

The research program was started 10 April 1965 and was completed 25 May 1965. USAAVLABS monitoring of the program was by Messrs. LeRoy Ludi and Andrew Connor. The Lockheed program was conducted by members of the helicopter staff under the direction of Mr. A. W. Turner, Flight Test Division Engineer.

Appreciation is due to USAAVLABS for their help in providing assistance and advice in planning and executing the entire research program.

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SYMBOLS

BUNO	Bureau of Naval Weapons Number (Abbr.)
b	wingspan
°C	temperature, degrees centigrade
C_D	drag coefficient, $C_D = \frac{Drag}{q\ S_W}$
C_L	lift coefficient, $C_L = \frac{Lift}{q\ S_W}$
$C_{L,W}$	wing lift coefficient
$C_{L_{max}}$	maximum attainable lift coefficient
C_{l_R}	rolling moment coefficient about the rotor mast $$C_{l_R} = \frac{l_R}{q\ S_W\ b}$$
C_{m_R}	pitching moment coefficient $$C_{m_R} = \frac{M_R}{q\ S_W\ \bar{c}}$$
C_m	longitudinal stability derivative, $\frac{dC_m}{d\alpha}$
C_Δ	auxiliary net thrust coefficient $$C_\Delta = \frac{F_N}{q\ S_W}$$
C_P	power coefficient, $HP/\rho\ (\pi R^2)\ (R\ \Omega)^3$
C_T	thrust coefficient, $W/\rho\ (\pi R^2)\ (R\ \Omega)^2$
$\bar{c}$	length of the mean aerodynamic chord
c.g., C.G.	center of gravity referenced to rotormast centerline
cps	cycles per second
Deg., ( )°	angular degrees
EAS	equivalent airspeed — $V_T\ \sigma^{1/2}$

ESHP	equivalent shaft horsepower
F_N	auxiliary net thrust
FRL	fuselage reference line, an arbitrary longitudinal line parallel to fore and aft centerline and waterline.
F_6	flapwise bending moment at rotor station 6
$F6_p$	helicopter pitching component of F_6
$F6_R$	helicopter rolling component of F_6
g	acceleration of gravity
HP	horsepower
IAS	indicated airspeed
i	incidence angle
i_t	tailplane incidence angle relative to FRL, + leading edge up
i_w	wing incidence angle relative to FRL, + leading edge up
K	constant term, defined where used
Kts	knots, nautical miles per hour
l_R	rolling moment about rotor mast
L_R	rotor lift
Lw	wing-body lift
M	Mach number
M_{hub}	moment produced by a blade acting on the hub
M_R	pitching moment about rotor mast

M_{shaft}	moment produced by entire rotor acting on the rotor mast
MAC	mean aerodynamic chord
mph	statute miles per hour
N. Dn	nose down
N. Up	nose up
n	flight load factor, multiples of g where noted, adjusted to a standard gross weight by the ratio $$n = n(\text{test}) \frac{\text{gross weight (test)}}{\text{gross weight (standard)}}$$
P	notation for per revolution when relating frequencies to rotor rotating frequency, e.g. 1P, one per revolution 3P, three per revolution
psi	pounds per square inch
psf	pounds per square foot
q	dynamic pressure $$q = \frac{\rho V_T^2}{2}$$
q_t	local dynamic pressure at the horizontal stabilizer
R	rotor radius
rpm	revolutions per minute
S	surface area
S_{HT}	horizontal tail area
S_w	wing area
SHP	shaft horsepower
THP	thrust horsepower = $F_N V_T/325$

V_c	calibrated airspeed
V_e	equivalent airspeed - $V_T \sigma^{1/2}$
V_i	indicated airspeed
V_L	design limit speed
V_{max}	maximum speed
V_T	true airspeed
W	weight
W_{AVG}	average weight
α	angle of attack
σ'	air density ratio $\sigma = \frac{\rho}{\rho_o}$
σ	rotor solidity
μ	tip speed ratio, $\frac{V \cos \alpha}{R\Omega}$
ρ	test condition air density
Ω	angular velocity, RAD/SEC
ρ_o	sea level standard day air density
θ_o	collective pitch at blade station zero, hub centerline, + leading edge up
$d\varepsilon/d\alpha$	rate of change of the wing downwash angle with respect to angle of attack

SUMMARY

This report covers the results of a speed extension program conducted by the Lockheed-California Company on the compound version of the rigid-rotor XH-51A helicopter (Figure 1) under contract DA 44-177-AMC-150(T), Modification 1. The principal objective of this program, which is a supplement to the program reported on in reference 3, was to investigate the flight characteristics of the compound helicopter with special emphasis on the areas of flying qualities, performance, structural loads, vibration, and maneuverability in the speed range of 200 to 230 knots true airspeed. The maneuvering envelope objective consisted of:

- 2.0g at 120 knots
- 1.75g at 200 knots
- 1.2g at 230 knots
- 0.8g at 230 knots
- 0.5g at 200 knots

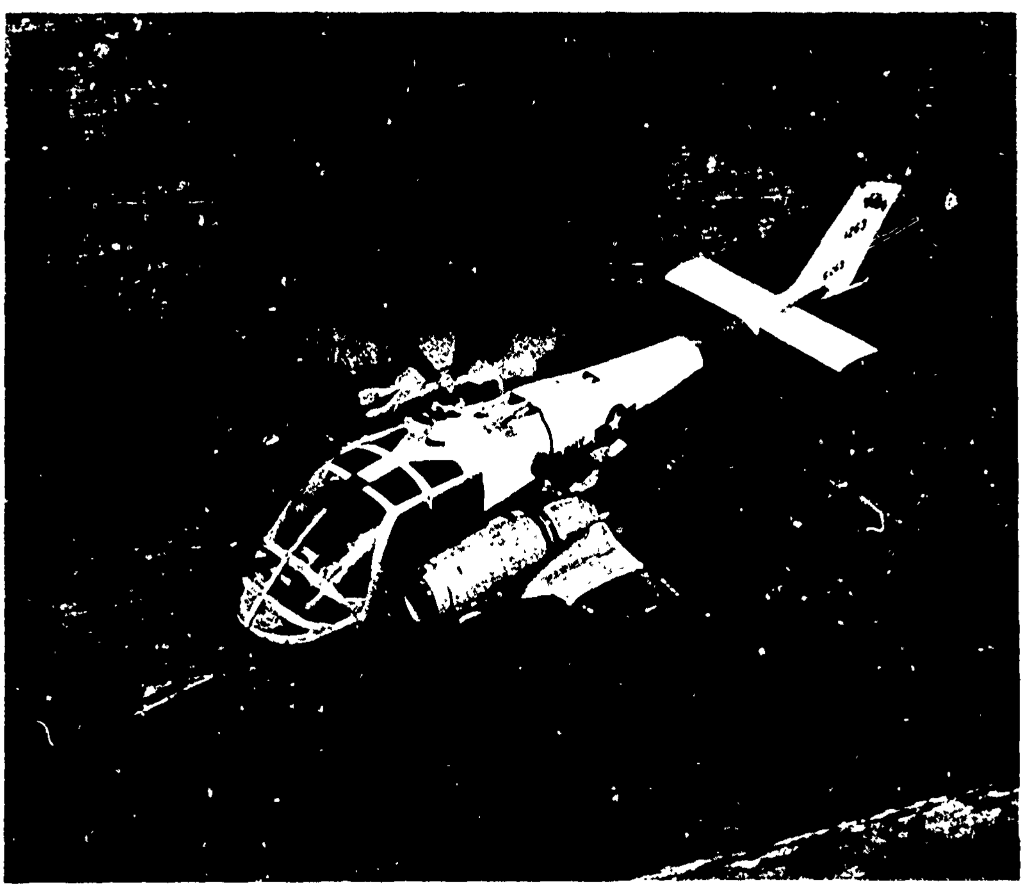

Figure 1. XH-51A Compound Helicopter In Flight

All contract objectives were met or exceeded. The maximum level flight true airspeeds demonstrated during this program were 236 knots at sea level and 228 knots at a density altitude of 12,000 feet. The maneuvering envelope investigated is shown in Figure 2. The shaded portion of this figure represents the current program test envelope, whereas the unshaded area reflects the results of the previous high-speed program conducted on this aircraft.

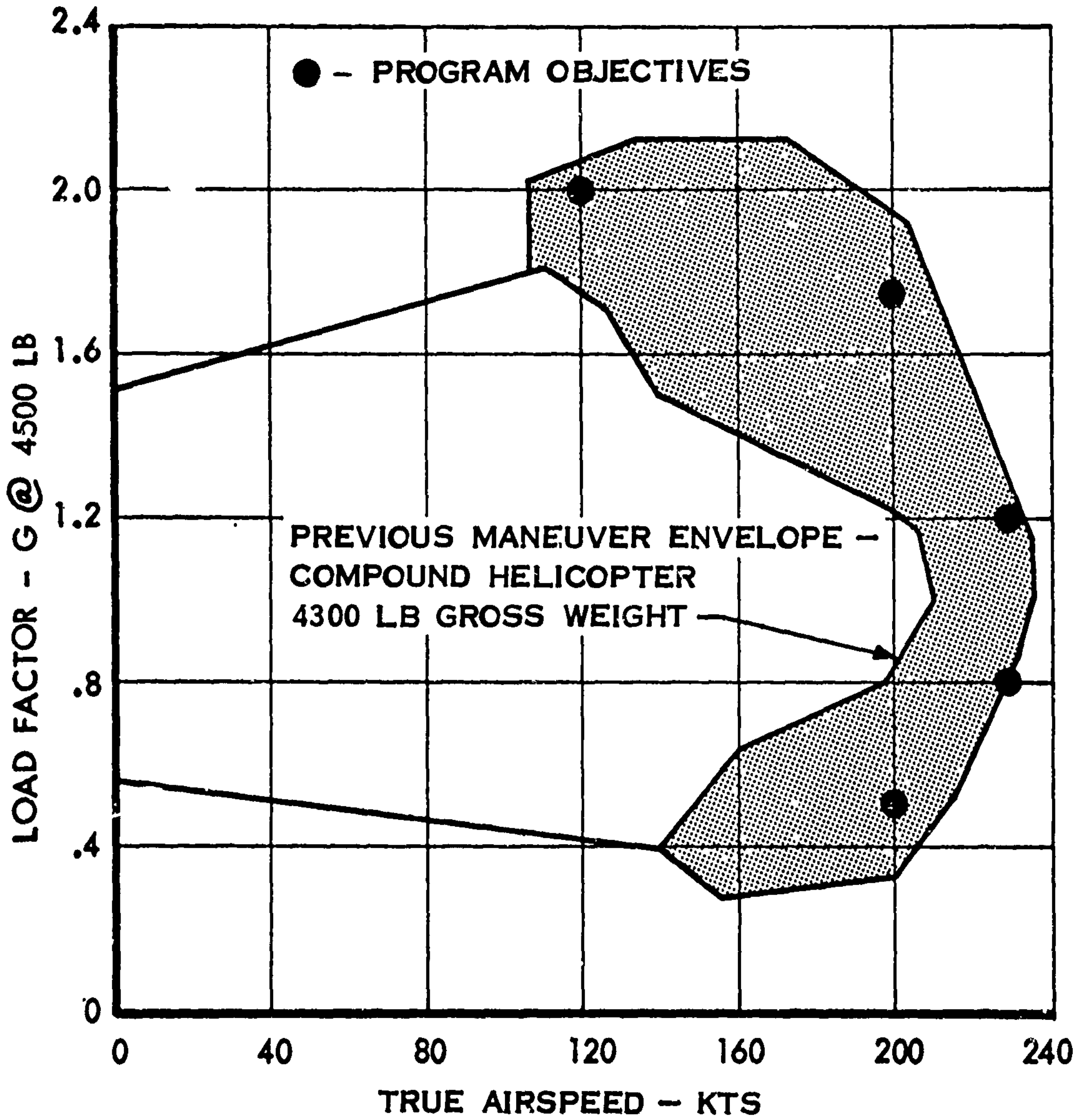

Figure 2. Maneuvering Envelope Versus True Airspeed

The only modifications to the helicopter accomplished since the completion of the previous flight program were:

- Hydraulic actuation of the wing spoilers,
- Reduction of span of the horizontal stabilizer from 11.0 to 9.0 feet (area reduced from 24.2 to 19.8 square feet),
- Tip weights and struts added to horizontal stabilizer,
- Additional extension added to the chord of the vertical fin,
- Additional windshield bracing and ram scoop to offset negative cabin pressure.

The major factor which limited further extension of the speed envelope during this program was a high level of vibration starting at a speed of approximately 220 knots and becoming more pronounced at higher speeds. Structural load measurements indicated that this vibration was caused by a marked rise in the rotor system higher order harmonic loads - particularly the three-per-revolution components. The overall structural loads, however, indicated only a gradual increase with speed and therefore were not a problem. The one-per-revolution component of pitch link loads indicated a similar pronounced increase concurrent with the start of the high vibration. This produced a strong lateral trim shift which, if much higher speeds had been attempted, would have been a limitation. The general level of flying qualities, other than this one item, remained satisfactory over the entire speed and maneuvering envelope.

The cause of this rapid increase in vibration, higher order loads, and trim requirements has been shown to be the result of high Mach numbers on the advancing blade. Each of these quantities shows a marked rise when the advancing tip reaches a Mach number of 0.9. High altitude tests were conducted to further evaluate this effect, and the results confirmed the low-altitude findings.

Although this factor limited the maximum speed of the current research program, a number of solutions are readily available and merit further investigation. The most direct and inexpensive of these is to simply reduce rotor rpm at high speed so that the onset of compressibility is delayed. Other methods, including the use of main rotor blades having a reduced second flap bending frequency or the tapering of blade thickness, are also available and are worthy of continued study.

Summarized in Table 1 are the boundaries of the flight envelope investigated during this program and the goals which were achieved.

TABLE 1 SUMMARY OF TEST CONDITIONS	
Maximum calibrated airspeed (Level flight)	220 Kts
Maximum true airspeed (Level flight)	236 Kts
Auxiliary thrust required at V_{max}	1,595 lb
Engine power required at V_{max}	245 HP
Rotor lift/gross weight at V_{max}	10%
Test pressure altitude at V_{max}	3,650 ft
Test density altitude at V_{max}	3,900 ft
Maximum calibrated airspeed at high altitude (Level flight)	190 Kts
Maximum true airspeed at high altitude (Level flight)	228 Kts
Auxiliary thrust required for V_{max} at high altitude	1,610 lb
Engine power required for V_{max} at high altitude	202 HP
Rotor lift/gross weight	3%
Test pressure altitude	10,200 ft
Test density altitude	12,000 ft
Maximum gross weight flown	4,800 lb
Maximum auxiliary thrust used in flight	1,850 lb
Minimum rotor lift/gross weight	0%
Maximum and minimum load factors	2.12 g at 133 and 172 Kts TAS 0.27 g at 156 Kts TAS 1.94 g at 200 Kts TAS 0.33 g at 200 Kts TAS
Maximum speed for autorotation	205 Kts TAS

CONCLUSIONS

1. The speed objective of 230 knots was exceeded with ample margins remaining in performance, flying qualities, and structural loads.

2. Vibration level was satisfactory at speeds up to 210 knots but became excessive as speeds approached 236 knots. This was the chief factor limiting further high-speed exploration and was attributed to high tip Mach numbers in combination with the relatively thick (12%) airfoil section of the blades.

3. A rather rapid increase in lateral trim requirement with increasing airspeed was noted commencing at approximately 215 knots. Although not a program limitation, this was felt to be a potential problem area which should be considered in future work.

4. Higher order harmonics of main rotor system structural loads - particularly the three-per-revolution component - showed a similar rapid increase at about 215 knots where tip Mach numbers exceeded 0.9.

5. Overall structural load measurements indicated a gradual rise with airspeed, but no significant problems in this regard were experienced.

6. Static longitudinal and maneuvering stability were found to remain positive throughout the flight envelope.

7. Longitudinal control became increasingly sensitive to pilot inputs as the speed exceeded approximately 180 knots. Other than producing a tendency for pilot-induced oscillations, this created no significant problem in meeting the program objectives.

8. Lift and thrust sharing between the main rotor and the auxiliary devices were investigated over a broad portion of the flight envelope. These results indicated that at the very high speeds, the performance tradeoffs favor the lowest collective setting consistent with wing lift capabilities.

9. Autorotation entries were conducted at speeds as high as 205 knots true airspeed with no difficulty. Rotor rpm was easily managed, and control response remained high.

10. Autogyro flight was found to be feasible and could be used for optimum site selection for autorotation landings.

RECOMMENDATIONS

The results of this program indicate a number of areas where additional study and flight research would prove beneficial in advancing the high-speed helicopter state of the art. These include:

1. Higher forward flight speeds appear entirely feasible. The next logical step, in this regard, would be to reduce the rotor rpm to approximately 90%. This reduced rpm would allow a true airspeed of approximately 250 knots to be reached before the tip of the advancing blade exceeds Mach 0.9. It was at this point that the blade loads, vibration, and trim requirements increased sharply, apparently due to the tip Mach effects.

2. Flight speeds above the currently limiting tip speed may be accomplished by installing main rotor blades which have a reduced second flap bending frequency. An experimental set of blades, which incorporate externally mounted tuning weights, have produced highly satisfactory results during recent testing of the conventional XH-51A helicopter. This modification would be particularly beneficial at the suggested lower rpm's since the blades have a significantly reduced response to three-per-revolution inputs.

3. The maneuvering capability of the compound helicpoter should be explored further. The results of the maneuverability investigations conducted to date have been very encouraging. Since no structural or other limits have been reached, this operationally oriented testing should be continued.

INTRODUCTION

Probably the most significant single factor which has prevented the helicopter from having a more widespread application is its inability to perform at high speed. Design complexity and lack of inherent stability are nearly as important from an operational standpoint and have also contributed to the helicopter's problems.

The rigid rotor design, under development at Lockheed since 1958, has largely eliminated the problems of rotor system complexity and lack of basic stability. To study the remaining important area - that of high-speed flight - the Lockheed-California Company, under contract to the United States Army Transportation Research Command*, Fort Eustis, Virginia, was authorized to modify one of the XH-51A helicopters, BUNO 151263, to a compound configuration. This modification consisted principally of adding a 70-square-foot wing and an auxiliary jet engine to the basic helicopter. No changes to the rotor system were required.

A research flight test program was conducted on this aircraft during the latter part of 1964. The objective of this program was to demonstrate the capabilities of the rigid rotor at flight speeds above 200 knots. As reported in Reference 2, this objective was met with no difficulty and a maximum flight speed of 210 knots was demonstrated.

The results of this program were very encouraging. Since no limitations had been encountered, Lockheed proposed that a speed extension program be authorized by the USAAVLABS to further explore the high-speed characteristics of the XH-51A with special emphasis on operational aspects, including maneuvering capabilities, structural loads, and vibration in the speed range between 200 and 230 knots. This research program was approved, and Modification 1 to Contract DA 44-177-AMC-150(T) was issued on 9 March 1965.

First flight under this contract extension was accomplished on 10 April 1965, and a maximum speed of 236 knots was demonstrated on 18 May 1965. During the program, 78 flights were conducted for a total of 15.3 flight hours. The flight program ended on 25 May 1965 with the last flight occurring on that date. The lack of any serious problems made it possible to complete this program 1½ months ahead of the contract schedule date.

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* Changed to U. S. Army Aviation Materiel Laboratories in March 1965.

DESCRIPTION OF THE TEST ARTICLE

At the completion of the previous compound helicopter program, certain modifications to the aircraft were considered to be necessary in order to extend the speed envelope further. Accordingly, prior to commencing the speed extension flight program, the following modifications were incorporated:

- Horizontal stabilizer area. The horizontal stabilizer was reduced in area from 24.2 to 19.8 square feet. This was accomplished by reducing the span 12 inches on each side. The reason for this change is discussed in the section on structural loads.

- Hydraulic actuation of spoilers. During the previous program the wing spoilers were cable-operated and once deployed could not be retracted in flight. Problems also existed in damping the opening load shock. To solve both problems, the spoiler mechanism was modified to a hydraulically actuated system. Manual control was retained as a safety backup.

During the course of the speed extension testing, additional modifications were incorporated which include the following:

- Weights and struts added to horizontal stabilizer. As discussed in the structural loads section, it was necessary to change the bending frequency of the horizontal stabilizer to reduce antisymmetric loads. This was accomplished by installing struts and tip-mounted weights.

- Vertical stabilizer tab. To reduce tail rotor loads at high asymmetric thrust levels, the vertical stabilizer tab was increased in area approximately 30%. This was accomplished by adding a 2.5-inch chord section to the upper 75% of the fin. This tab was less than full-span to prevent interference with the tail rotor at its teetering limits. External braces were added to the tab to assist in maintaining the 20 degree deflection.

- Windshield braces and ram scoop. Additional internal bracing of the windshield became necessary at the higher flight speeds of this program. The windshield problem was due to high dynamic pressure in combination with lower-than-ambient cabin pressure. The latter problem was partially solved by the installation of a ram scoop designed to supply ram air to the transmission compartment.

Most of the preceding changes are visible in the photograph of Figure 3 below. Details of the compound helicopter description are listed in Table 2.

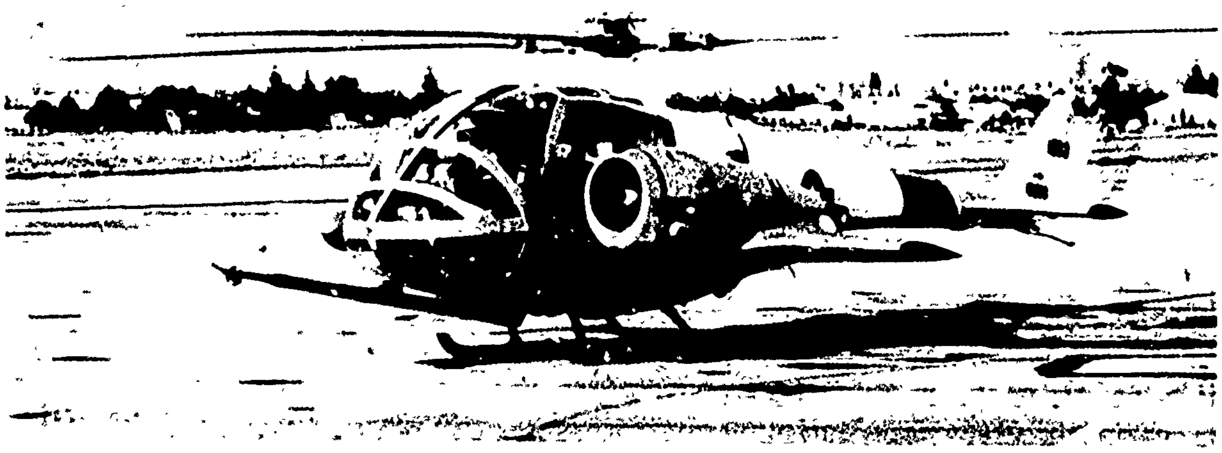


Figure 3. XH-51A Compound Helicopter on the Ground

TABLE 2
COMPOUND HELICOPTER DESCRIPTION

General	
Design gross weight	4,500 lb
Fuel capacity	475 lb
Normal crew (plus research instrumentation)	2
Overall length	42.58 ft
Maximum ground attitude (tail low)	6°
Roll mass moment of inertia (including rotor)	1,500 slug-ft^2
Pitch mass moment of inertia (including rotor)	3,180 slug-ft^2
Yaw mass moment of inertia (including rotor)	3,800 slug-ft^2
Main Rotor	
Type	rigid
Diameter	35 ft
Number of blades	4
Blade chord	13.5 in
Blade weight	86 lb/blade
Airfoil section	modified NACA 0012
Blade taper	0
Blade twist (root to tip)	-5°
Rotational axes tilt	6° forward
Hub precone	+3.2°
Preset blade droop @ sta. 27.85	-1°
Disc area	962 ft^2
Solidity	.0818
Disc loading	4.68 lb/ft^2
Polar moment of inertia	1,013 slug-ft^2
Normal operating speed	355 rpm
Blade sweep	1.4° forward

TABLE 2 (cont'd)	
<u>Control Gyro</u>	
Diameter	72 in
Number of arms	4
Polar moment of inertia	7.5 slug-ft^2
Incidence angle of arms	5.0°
<u>Tail Rotor</u>	
Diameter	72 in
Number of blades	2
Blade chord	8.5 in
Hub type	teetering
Airfoil section	NACA 0012
Blade taper	0
Blade twist (root to tip)	-4.35°
Feathering moment balance weights:	
weight	2.25 lb/blade
arm	3.0 in
Delta -3 hinge	15°
Disc area	28.27 ft^2
Solidity	.1503
Pitch change travel	27° to -8°
Normal operating speed	2,085 rpm
<u>Wing</u>	
Span (nominal)	16.83 ft
Taper ratio	.5
Area	70 ft^2
Aspect ratio	4.05
Sweepback (.25c)	0
Chord (MAC)	51.72 in
Airfoil	NACA 23012
Incidence (fixed)	-.9°

TABLE 2 (cont'd)	
<u>Horizontal Stabilizer</u>	
Span	108 in
Chord (constant)	26.4 in
Area	19.8 ft^2
Aspect ratio	4.1
Incidence	-0.25°
Airfoil section	NACA 0015
Tip weights	8 lb/side
<u>Vertical Stabilizer</u>	
Span	41.75 in
Chord (tip)	38.5 in
Chord (root)	51.5 in
Area	12.68 ft^2
Taper ratio	.70
Aspect ratio	.95
Airfoil section	modified NACA 4424
<u>Powerplants</u>	
Primary	
Type	Turboshaft
Maximum power (takeoff)	500 SHP @ sea level
MIL power (30 minute limit)	450 SHP @ sea level
Fuel type	JP-4
Oil type	Turbo 35
Auxiliary	
Type	Turbojet
Military thrust @ 200K	2,490 lb @ sea level
Fuel type	JP-4
Oil type	Turbo 35

RESEARCH FLIGHT TESTS

This section of the report covers the results of the research flight tests conducted during the speed extension program. Inasmuch as this report is concerned primarily with high-speed data, low-speed data have not been included on many of the figures.

The information in this section is presented in the following sequence:

1. Performance
2. Flying Qualities
3. Structural Loads
4. Vibration
5. Pilot Observations
6. Problems Encountered and Solutions

PERFORMANCE

Airspeed Calibration

The boom airspeed system was calibrated by the pacer aircraft method at calibrated airspeeds up to 187 knots during the previous test program. At the higher speeds scheduled for the present program, however, it was not feasible to continue airspeed calibrations in this manner due to speed limitations of the pacer aircraft, and the possibility of incurring large errors in high-speed flight.

Accordingly, a direct and more accurate airspeed calibration was accomplished by the altimeter depression method at speeds up to 193 knots calibrated airspeed. These results indicated that although the differences from the previous calibration were small, a more conservative fairing of the high-speed data would give added validity to the extrapolated portion of the curve. The modified airspeed calibration is shown in Figure 4.

Level Flight Performance

The compound helicopter represents a step forward in the current state of helicopter technology. Compared with a pure helicopter, the compound is somewhat more complex because of its added sources of lift and propulsion. This added complexity introduces new variables into the analysis of level flight performance data which are not easily handled by the conventional forms of helicopter performance parameters (C_T, C_P, etc.). One of the implied objectives of this program was to study and develop new techniques which would adequately define the many variables involved in the analysis of compound helicopter performance. The procedures which have been found to provide satisfactory levels of accuracy are as follows:

Power Required. From basic aerodynamic theory, the power required for level flight is uniquely defined in terms of a C_L-C_D relationship. For the compound helicopter these parameters may be defined as follows:

$$C_L = \frac{W}{q\,S_w} = \frac{W}{\frac{1}{2}\rho V_T^2\, S_w}$$

$$= \frac{W}{\rho(\pi R^2)(R\Omega)^2\left(\frac{V_T}{R\Omega}\right)^2} \cdot \frac{\pi R^2}{\frac{1}{2}S_w} = \frac{C_T}{\mu^2} \cdot \frac{\pi R^2}{\frac{1}{2}S_w}$$

$$C_D = \frac{\text{ESHP}\,(550)}{q\,S_w\,V_T} = \frac{\text{ESHP}\,(550)}{\frac{1}{2}\rho V_T^3\, S_w}$$

$$= \frac{\text{ESHP}\,(550)}{\rho(\pi R^2)(R\Omega)^3\left(\frac{V_T}{R\Omega}\right)^3} \cdot \frac{\pi R^2}{\frac{1}{2}S_w} = \frac{C_P}{\mu^3} \cdot \frac{\pi R^2}{\frac{1}{2}S_w}$$

By definition, equivalent shaft horsepower in the above expression is equal to the sum of the engine shaft horsepower and the auxiliary thrust expressed in terms of thrust horsepower:

$$\text{ESHP} = \text{SHP} + \frac{F_n V_T}{325}$$

where V_T is expressed in knots, and the other terms are defined in the symbols section of this report.

For a given rotor rpm and W/σ, C_L varies only with airspeed, while C_D varies with airspeed and ESHP/σ. This somewhat simplifies the computational procedure since the level flight performance data can be plotted in terms of ESHP/σ versus true airspeed. This has been accomplished, as shown in Figure 5, for a gross weight to density ratio of 4500 pounds. As expected, this produced a single fairing for all test points.

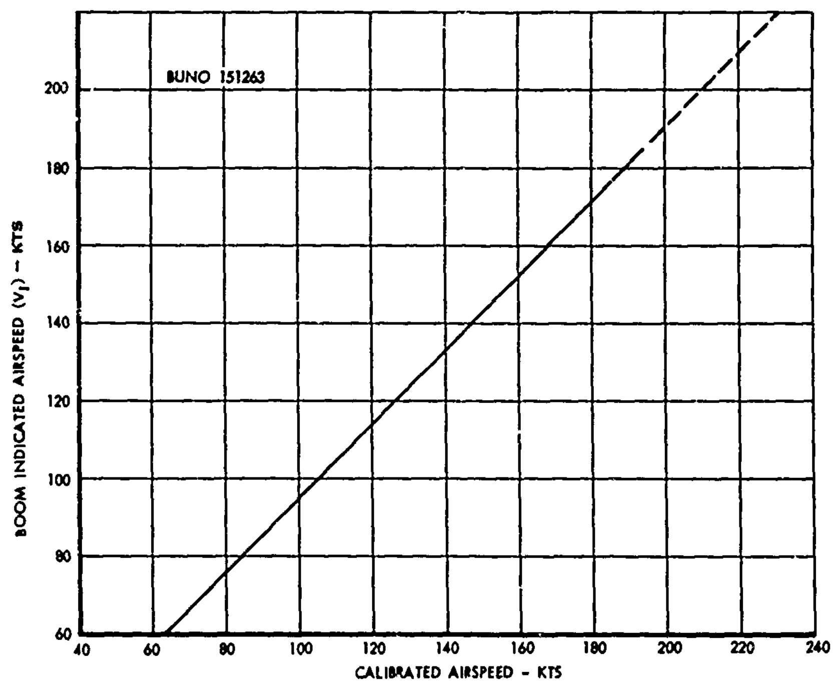


Figure 4. Airspeed Calibration - Boom System

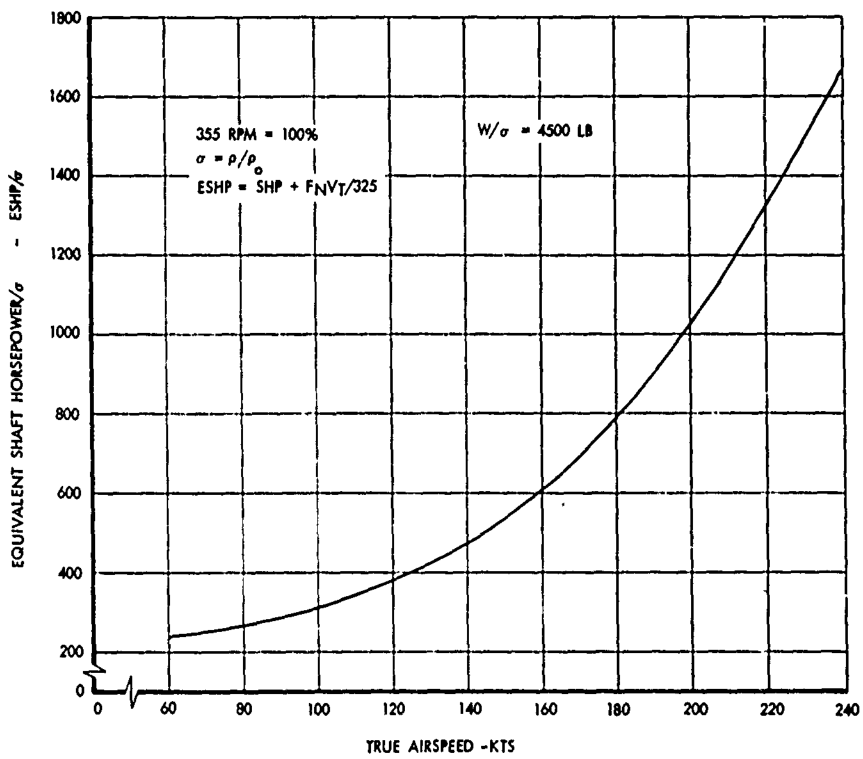


Figure 5. Equivalent Shaft Horsepower Versus True Airspeed - Level Flight

The tradeoff between shaft horsepower and auxiliary thrust is easily determined from this figure by assuming various levels of thrust and then deducting the thrust horsepower from the equivalent shaft horsepower. For example, assume a sea level, standard day, and

$$W/\sigma = 4{,}500,\ V_T = 170 \text{ Kts},\ ESHP = 690\ \ F_N = 1{,}000 \text{ lb};$$

then,

$$SHP = ESHP - \frac{F_N V_T}{325}$$

$$= 690 - \frac{(1{,}000)(170)}{325} = 167$$

Using this procedure, the variation of shaft horsepower with auxiliary thrust was computed over the forward flight envelope and is shown in Figure 6.

Rotor Lift-Power Sharing. The amount of lift supplied by the rotor for a given flight condition is a function of shaft horsepower, auxiliary thrust and airspeed. Figure 7 represents the flight test derived variation of rotor lift-to-weight ratio as a function of power fraction, where

$$\text{Power Fraction} = \frac{SHP}{ESHP}$$

This method of presentation was also found to provide a single fairing for all test data.

Using these data, together with the information included in Figure 6 the rotor lift-to-weight ratio for a full spectrum of shaft horsepower and auxiliary thrust have been computed. This information is shown in Figure 8.

Thrust Sharing. The data shown on the lower half of Figure 9 indicate the variation of body axis angle of attack with airspeed for a number of fixed collective pitch settings. These data are valid only for level flight and show quite clearly the attitude changes which are necessary to maintain a constant total lift on the aircraft for various rotor lift conditions.

An interesting characteristic is evident from Figure 9. At low speeds, when the collective pitch is increased 1 degree, the body attitude decreases nearly 2 degrees. At high speed, however, a 1-degree change in collective pitch results in less than 1/2-degree change in body attitude. Thus, at low speed, increased collective pitch actually produces a lower angle of attack on the advancing blade, whereas at high speed increased collective pitch results in an increased

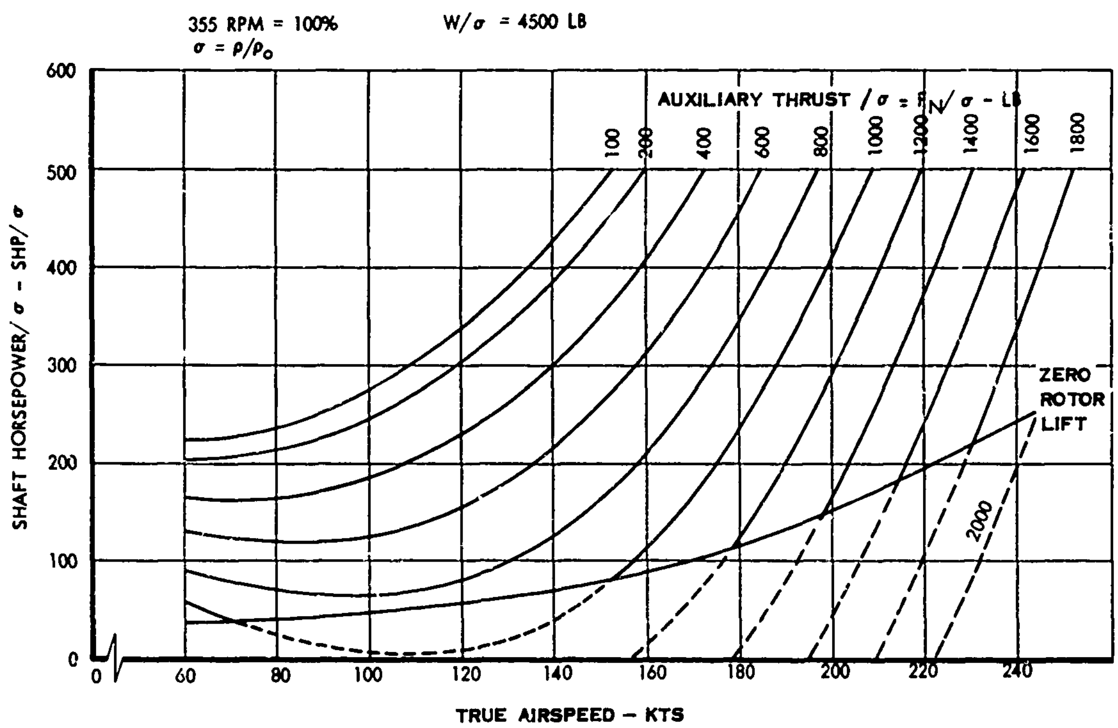


Figure 6. Variation of Shaft Horsepower With Auxiliary Thrust - Level Flight

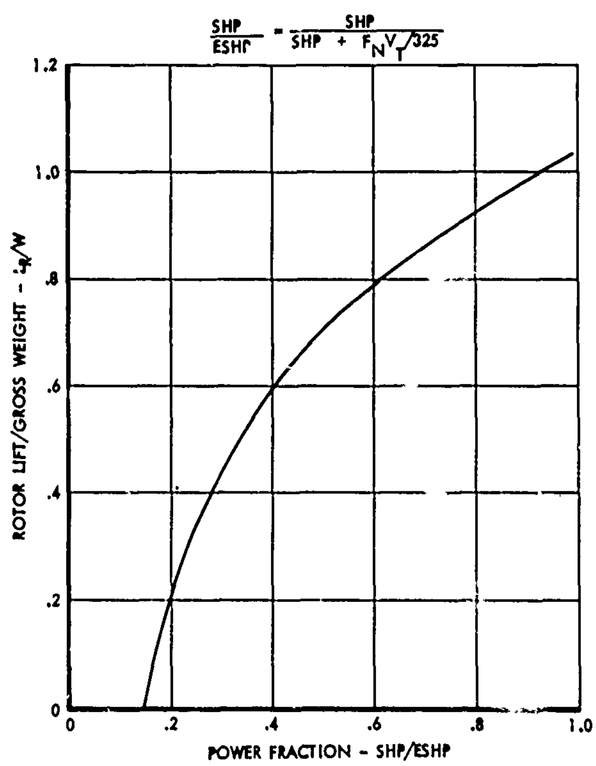


Figure 7. The Effect of Rotor Lift-To-Weight Ratio on Power Fraction

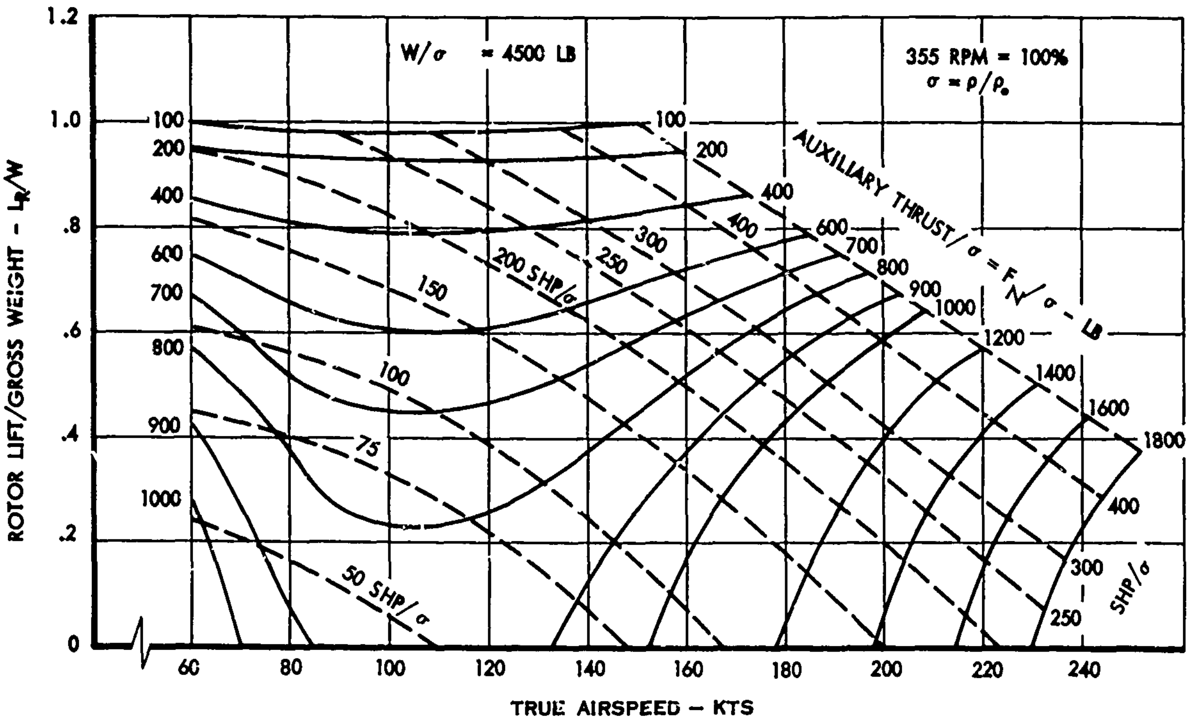


Figure 8. Variation of Rotor Lift With Power and Thrust - Level Flight

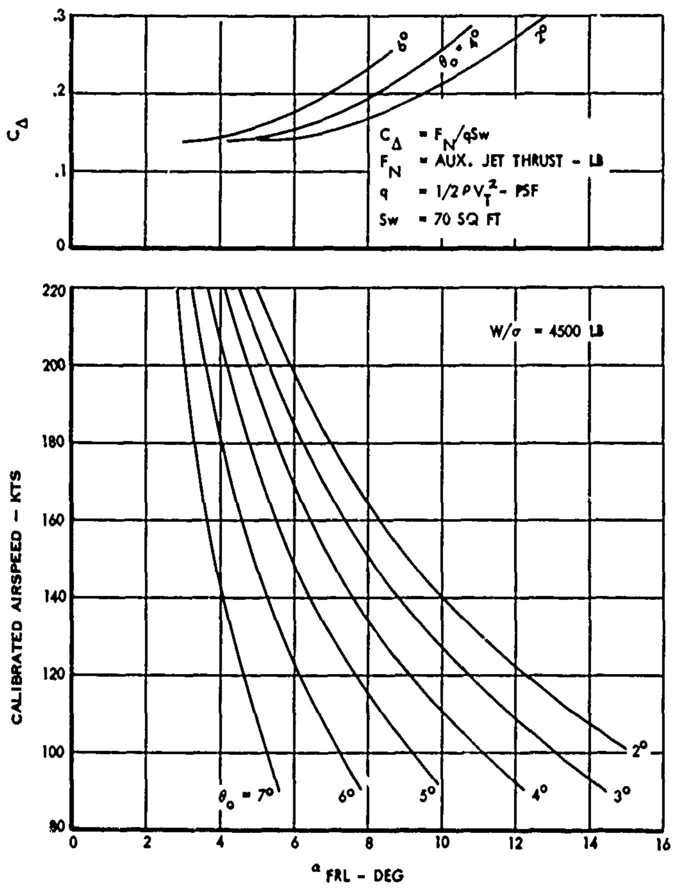


Figure 9. Body Attitudes and Thrust Coefficient
Versus Collective Pitch

blade angle of attack. This may explain the crossover of the lateral cyclic control motion data at approximately 100 knots as shown in Figure 14.

Shown at the top of Figure 9 is the auxiliary thrust required for level flight in terms of thrust coefficient for a number of constant collective pitch settings. These data tend to confirm that the level flight performance of the compound helicopter generalizes quite well into a unique C_L-C_D polar. This can be seen by noting that this data presentation is actually a plot of C_D versus α , since C_D is identically equal to C_Δ if the rotor thrust effects - in this case, shown as variations in θ_0 - are considered to merely bias the total drag of the aircraft.

The information on this figure has been used to compute the auxiliary thrust requirements of the helicopter in level flight for a number of rotor lift conditions. These data are shown in Figure 10 for a gross weight-to-density ratio of 4,500 pounds. It is apparent from this figure that for speeds in excess of about 160 knots, the improvement in auxiliary thrust required with increased collective pitch diminishes rapidly. At speeds below 160 knots, however, higher rotor lift pays significant dividends. This is not only due to the lift augmentation but also to the fact that unless the higher settings are used, the rotor actually produces a large drag component rather than forward thrust because of adverse body attitudes.

For the XH-51A compound, there appears to be no finite point where immediate transition from a high to a low collective setting would produce optimum results. Rather, the data indicate that economics, vibration, and other factors tend to favor near-zero rotor lift at high speed, with progressively more rotor lift becoming beneficial as the speeds approach those of the conventional helicopter.

Climb Performance

One area where increased rotor lift is beneficial is during high performance climbs. The optimum climb speed on the XH-51A compound is on the order of 120 knots calibrated airspeed. At this speed, moderate rotor thrust in combination with normal rated thrust on the auxiliary engine produced rates of climb in excess of 3,500 feet per minute. On three occasions, climbs from liftoff to 10,000 feet were accomplished in less than 5 minutes. In view of the 4,600-pound average weight of the helicopter, this is considered to be an excellent level of performance.

Hover Performance

A stable hover point in the pure helicopter mode was obtained at a skid height of 4.5 feet. At this condition the power required, corrected to standard day conditions, was 413 shaft horsepower at a gross weight of 4,525 pounds. No attempt was made to gather additional data since this was not one of the major program objectives.

Wing-Body Lift Characteristics

Figure 11 shows the wing-body lift characteristics of the helicopter as a function of angle of attack. The lower line on this figure defines the lift characteristics with the spoilers extended. These lift data, which are plotted in conventional coefficient form, were obtained from strain gauges attached to the four transmission support links. These measurements were calibrated up to 5,000 pounds in the laboratory and were checked by suspending the helicopter and using a standard load cell. This method has been found to provide a highly accurate indication of auxiliary lift, with good correlation being evident throughout the level flight and maneuvering envelope.

Techniques of Analysis

The performance analysis methods described in the preceding sections of this report are unique to the compound helicopter and this program. Although they have provided consistent results in nearly every case, they are based on a relatively small data sample. For this reason, caution in their use is suggested until a broader application substantiates their validity. Work in this respect is continuing, and improved methods will undoubtedly be developed as more compound helicopter experience is gained. The important point in presenting these preliminary methods at this time is to show that simple analysis methods appear to be feasible and to encourage further effort in this important area.

FLYING QUALITIES

One of the principal objectives of this program was to further explore the flying and handling qualities of the rigid rotor compound helicopter in the high-speed environment above 200 knots. In meeting this objective, tests were conducted up to a maximum speed of 236 knots without encountering any serious problems insofar as flying qualities were concerned. The first indications of adverse trim requirements due to advancing blade compressibility effects were observed, however, which indicated that new techniques or flight procedures would be required if speeds much above this figure were to be investigated.

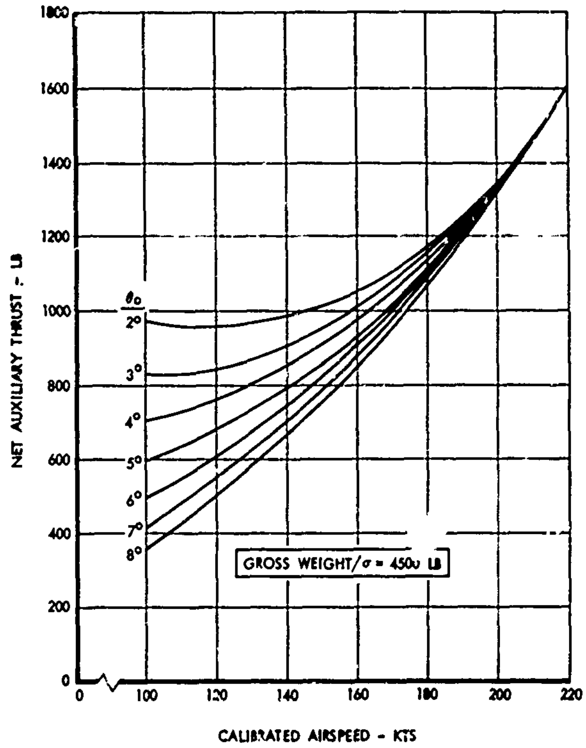


Figure 10. Auxiliary Jet Thrust Required For Various Collective Pitch Settings - Level Flight

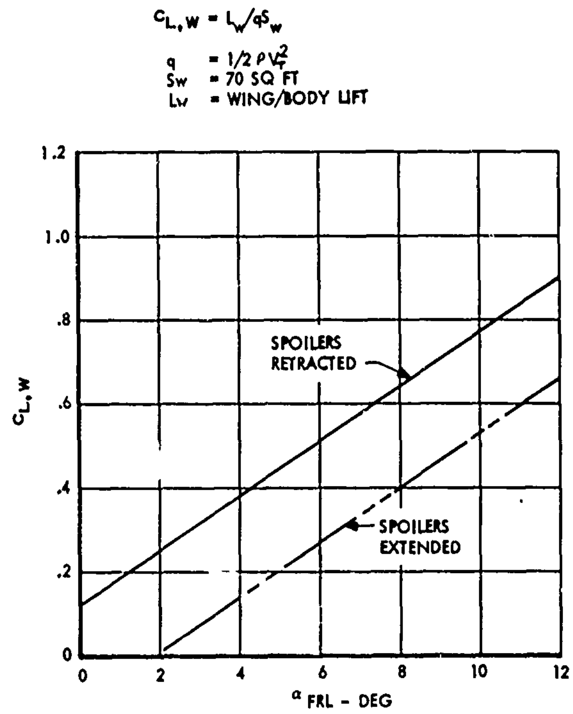


Figure 11. W[illegible]ng-Body Lift Characteristics

The flying qualities tests which were performed during this program extension consisted mainly of static longitudinal and lateral stability, trim characteristics, control power, control response, longitudinal damping, maneuvering stability, and autorotation entries. In every area, the rigid rotor was found to have the characteristics which are required for sustained operation above 200 knots.

Static Longitudinal Stability

Because of vibration and structural load considerations, the horizontal stabilizer was reduced in area from 24.2 to 19.8 square feet prior to the start of this program. Longitudinal stability data with the new stabilizer are shown in Figure 12. These data indicate that the stability is positive and confirm the anticipated change due to the reduction in tail area. An additional effect which had also been predicted is the reduced trim shift due to rotor downwash as shown on the lower half of this figure. (For a discussion of this effect, see reference 2.)

During the flight testing of the compound helicopter, three different horizontal tail sizes were evaluated together with a number of incidence angles for each stabilizer. Results from these tests have been used to compute the important longitudinal stability parameters for comparison with wind tunnel results. This comparison is tabulated below and indicates that excellent correlation exists between flight-measured and wind tunnel data.

TABLE 3

LONGITUDINAL STABILITY COMPARISON

Stability Term	Flight-Measured	Wind Tunnel
Tail efficiency, q_t/q	0.74	0.77
Downwash derivative, $d\varepsilon/d\alpha$	0.65	0.58
Tail contribution to stability,	-0.020	-0.025

Although Figure 12 indicates that the longitudinal stability is positive, the pilot reported that the longitudinal cyclic control motion was not in the proper direction as speed increased. This apparent instability is actually the result of trim changes associated with rotor downwash and auxiliary thrust changes and is discussed in the section below on trim characteristics.

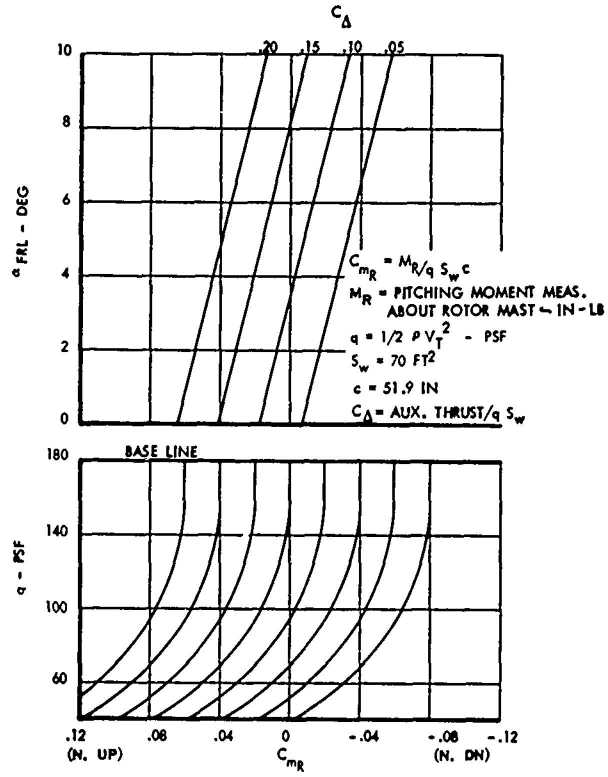


Figure 12. Static Longitudinal Stability and Trim Requirements

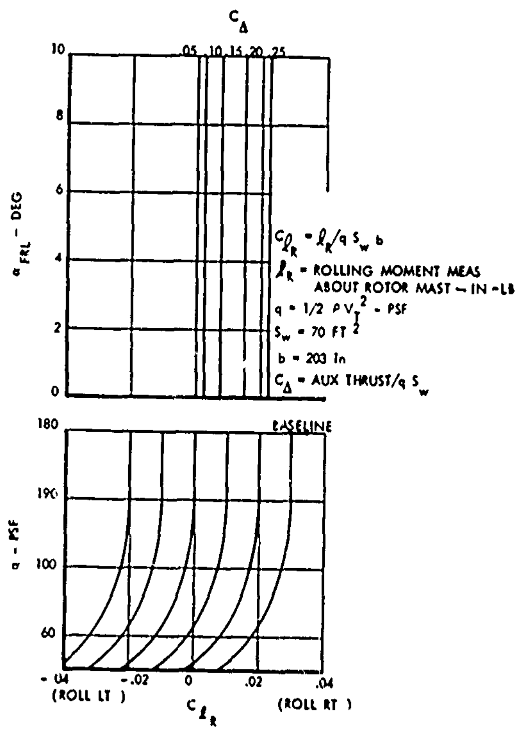


Figure 13. Lateral Trim Requirements

Static Lateral Stability

Static lateral stability remains essentially unchanged from the previous test results. The test data, in this regard, indicate that both lateral stability and damping are light but positive. This was confirmed by pilot qualitative observations.

Lateral trim, on the other hand, was found to exhibit a new trend. This characteristic is discussed at some length in the section on trim characteristics below. The wing and body contribution to lateral trim is shown on Figure 13.

Trim Characteristics

Trim requirements over the speed envelope are shown in Figure 14 in terms of longitudinal and lateral cyclic control motions. These data indicate that not only is there a shift in cyclic control with collective pitch but that the gradual slope of control motion with airspeed for the lateral cyclic control has a pronounced break at a calibrated airspeed of approximately 200 knots. As discussed below, this break is probably caused by compressibility effects on the advancing blade. The reason that the break is more evident in the lateral cyclic control than in the longitudinal will also be discussed.

As previously noted, the pilot reported that as speed increased the cyclic control tended to move aft. This is reflected in the data and occurs as a result of auxiliary thrust and rotor downwash influences. The pilot also reported, however, that increasingly more right cyclic control was required as speed increased. The data shown in Figure 13 tend to contradict this since it indicates that the body moments produce a slight right rolling tendency as speed increases.

The explanation for this apparent discrepancy is that feathering moments are produced as a result of loads in the rotor system which cause the control gyro to react. These moments must, therefore, be balanced by cyclic control inputs to maintain the helicopter in trim. To understand how this occurs, it is first necessary to note that the rotor hub is exactly the same as that used for the basic XH-51A and has a precone angle of 3.2 degrees. At low collective pitch settings where the rotor lift is less than about 4,000 pounds, the rotor is substantially under-coned. Thus, as the drag of the advancing blade increases with increased flight speed, a one-per-revolution feathering moment is produced which tends to reduce the blade's pitch. This feathering moment, transmitted through the pitch links, causes a precessional moment on the gyro. The sketch shown in Figure 15 indicates how this takes place and also shows the control phasing required to balance these moments.

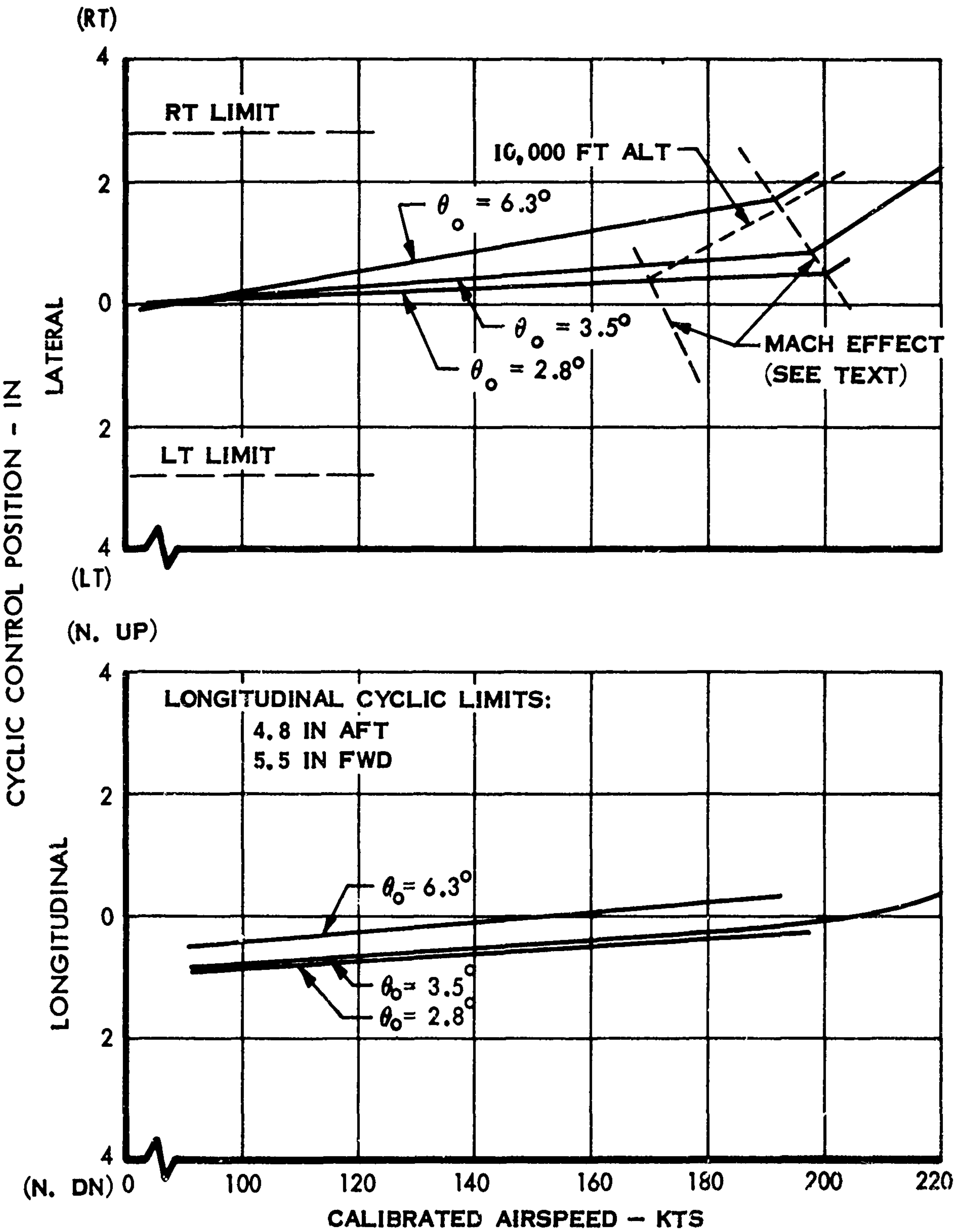


Figure 14. Cyclic Control Motion Versus Airspeed

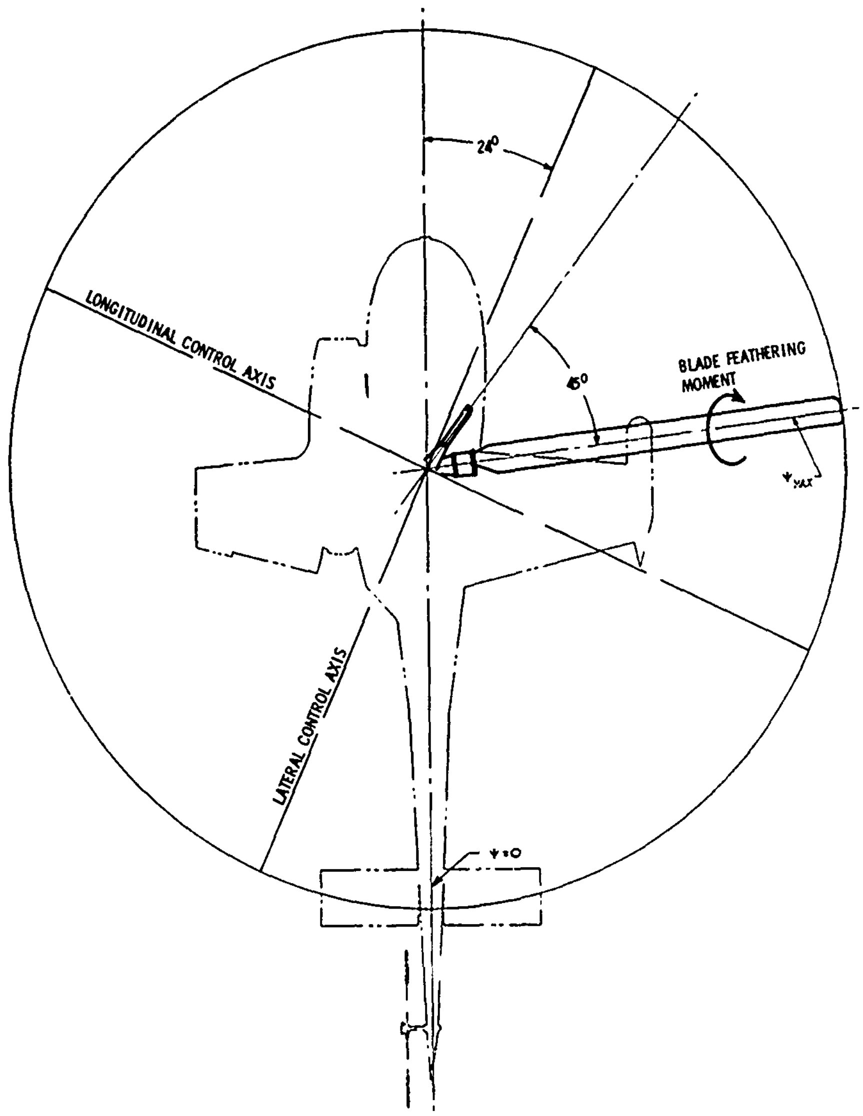


Figure 15. Phase Relationship Between Blade Feathering Moments and Control System

The data shown in Figure 16 confirms the presence of this one-per-revolution precessional moment in terms of measured loads in the pitch links. This figure further shows that the blade azimuth angle where the pitch link is in maximum tension averages about 110 degrees (from straight aft). Considering that the gyro leads the blade by 45 degrees, and that the control axis is phased 24 degrees (anti-rotation) from the body axis, as shown in Figure 15, this one-per-revolution moment is manifested almost entirely as a lateral cyclic control requirements. This explains the greater sensitivity of the lateral cyclic control to both collective pitch inputs as well as increasing speed.

The pronounced break in the lateral cyclic control motion plot at approximately 200 knots appears to be the result of compressibility drag rise brought about by high tip Mach number. Notice that the pitch link loads show this same effect. When plotted against tip Mach number, these loads show a consistent break at Mach 0.9. Carpenter, in reference 3, shows that a 12% thick rotor blade at low lift ($\alpha = 0°$) starts to experience compressibility drag rise at Mach 0.8. Although he points out that this is based on hovering flight, his data indicate that tip Mach effects are strong and should be expected at advancing tip Mach numbers near 0.8, even though the rotor is partially or totally unloaded. The pronounced reduction in critical Mach number with rotor lift, or angle of attack (shown in reference 3), gives clear evidence here of the benefits to be derived from rotor unloading at high forward flight speeds.

To further explore this potential problem area, two high-speed tests were conducted at a pressure altitude of 10,000 feet. As shown in Figures 14 and 16, the compressibility problem was somewhat magnified and further high-altitude, high-speed testing was not attempted. It is interesting to note that again the break in the pitch link loads and cyclic control motion plots occurred at a tip Mach number of 0.9. This tends to confirm that the characteristic trend of cyclic control motion with speed is associated directly with advancing blade drag rise and not with some other phenomenon.

In its present configuration, and using the techniques of this program, it is evident that further extension of the speed envelope would be limited by lateral cyclic control requirements. The simplest and most direct means of dealing with this problem, however, is to utilize reduced rotor rpm. This would probably have some effect on retreating blade stall, but since this was not found to be a problem, the possible advantages of reduced tip speed would appear to be worthy of further study and investigation.

Control Power

Cyclic control power, in terms of pitch and roll rate per inch of cyclic control input, was investigated over the speed envelope.

Longitudinal control power, as shown on Figure 17, increases with flight speed. At speeds less than 180 knots, the response is comfortable and easily managed. At higher speeds, however, excessive load factors tend to occur when cyclic inputs are made unless special piloting precautions are observed. This is not only due to the increased cyclic sensitivity but also to the fact that load factor varies as the product of pitch rate and airspeed which tends to magnify the response of the helicopter to control inputs. This same characteristic has been found more or less universally in both pure helicopters as well as in fixed-wing

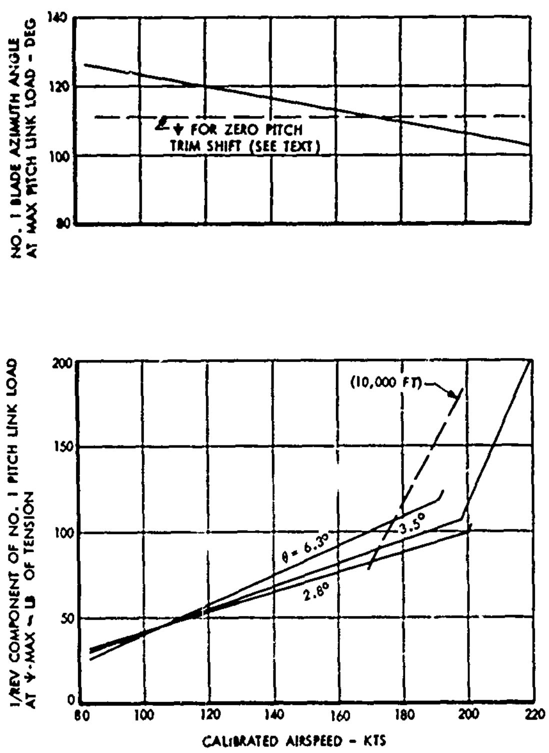


Figure 16. One Revolution Component of Pitch Link Axial Load

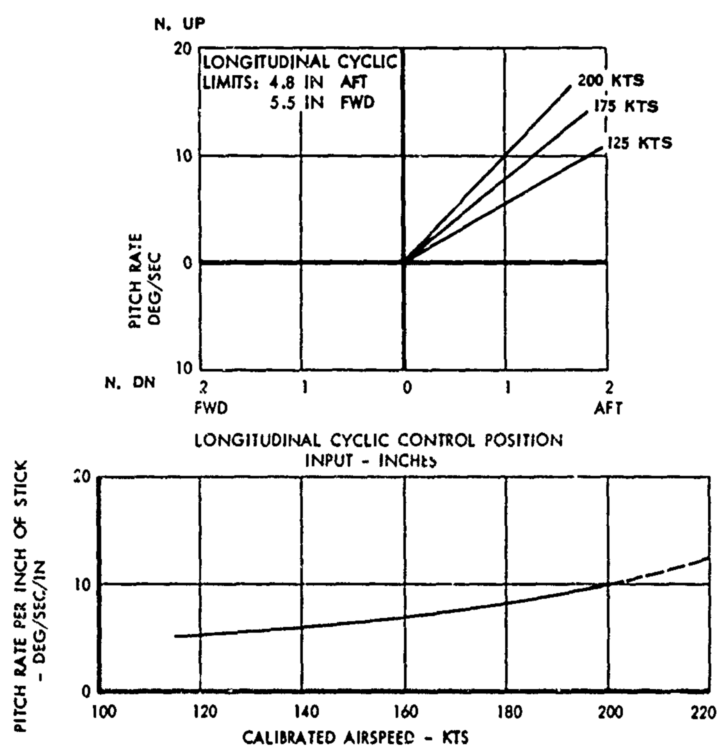


Figure 17. Longitudinal Control Power

aircraft. The problem here is not great, although it does indicate that future compound helicopters will almost certainly require some means of adjusting control system sensitivity to match the flight environment.

Lateral control power tests were conducted over the same envelope as the longitudinal testing. Unfortunately, the roll rate gyro malfunctioned and could not be used to provide a quantitative measure of control power. Other methods of determining this quantity were attempted, but none provided usable results. A reasonable evaluation of this characteristic, however, can be obtained intuitively from previous experience and compared with pilot observations for confirmation. The steady-state roll rate per inch of cyclic control input of the pure helicopter version of the XH-51A is approximately 12 degrees per second. Since the higher roll inertia of the compound only affects roll acceleration, its steady-state roll rate should be equal to the response of the pure helicopter reduced by the effect of wing-induced roll damping. This has been conservatively calculated to produce approximately a 35% reduction in roll rate. Thus, a rate of 7-8 degrees per second per inch of lateral cyclic input would be expected. The pilot reported something less than this based on his qualitative observations. This may be the result of increased control system lag and lower roll acceleration which exists due to the higher roll inertia.

The pilot also reported a marked difference in roll response between left and right rolls. None of the test data, which was admittedly questionable in this area, indicated this trend, nor does a rational analysis of the forces and moments acting on the helicopter. One explanation for this is the large lateral separation between the pilot and the roll center of the aircraft which is some 3 inches to the left of the centerline. This could easily produce the sensation reported by the pilot. Until additional test data are obtained, this area will remain unresolved.

Control Response and Short Period Damping

Longitudinal control responses to step inputs for true airspeeds of 125, 190, and 211 knots are shown in Figures 18, 19, and 20. These data indicate that the helicopter responds to a cyclic input in approximately 0.2 second. Within approximately 1 second, the point of inflection in the vertical acceleration trace occurs. These characteristics compare favorably with fixed-wing aircraft response and further satisfy the helicopter requirements specified in MIL-H-8501A in this regard.

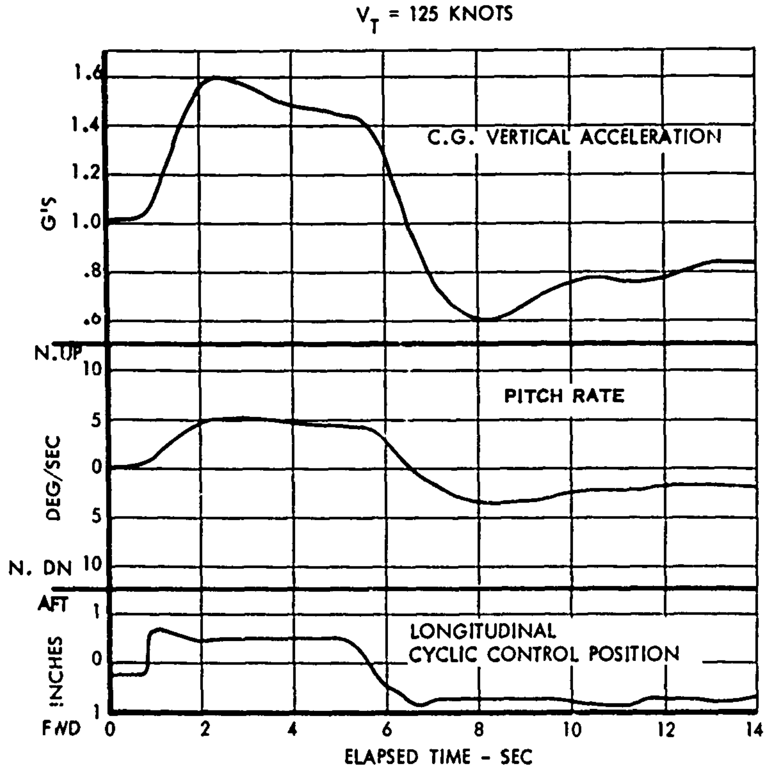


Figure 18. Time History of Longitudinal Control Power - 125 Knots

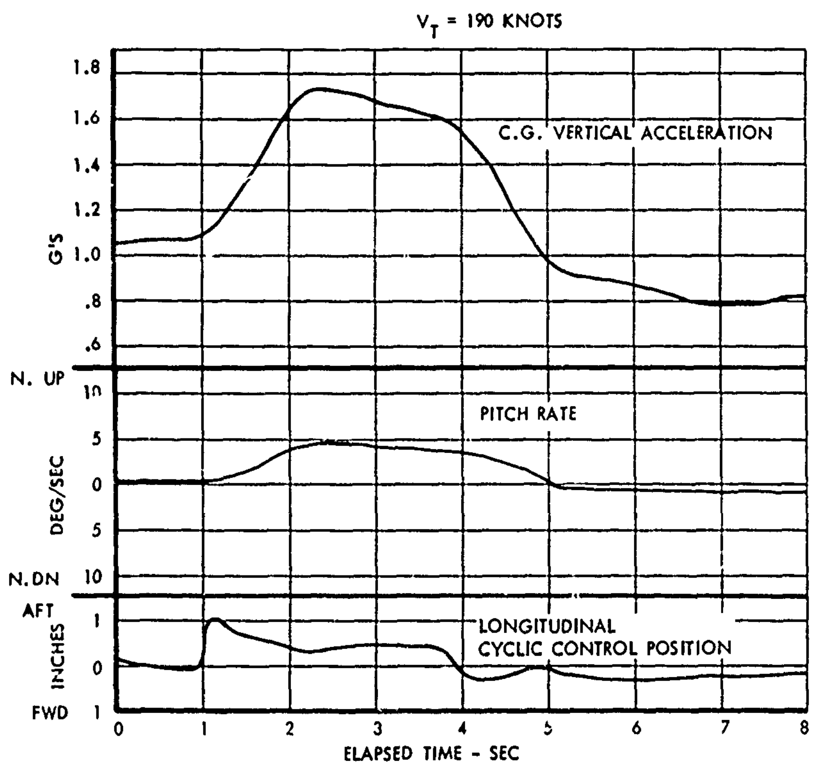


Figure 19. Time History of Longitudinal Control Power - 190 Knots

Pulse inputs to investigate short period damping and control response are shown in Figures 21, 22, and 23 at true speeds of 140, 180, and 207 knots, respectively. These data indicate that the short period is heavily damped, requiring less than 1-1/2 cycles to damp completely. The period is approximately 3 seconds. Pilot observations of this characteristic confirm the strong damping in pitch. Control response from these tests agree with the results discussed above for control step inputs.

Long Period Damping

The characteristics of the phugoid were investigated over the flight envelope up to calibrated airspeeds of 200 knots. The results of these tests were not conclusive since trim difficulties, weather conditions, and the high-speed longitudinal sensitivity discussed above prevented any full cycle data from being obtained. Qualitative observations, however, indicated that phugoid damping was neutral to positive and that the period was on the order of 20 seconds. In this respect, the long period characteristics are considered satisfactory for high-speed operation.

Maneuvering Stability

Maneuvering stability, in terms of control forces required to produce normal load factors, was investigated over a flight envelope from 100 to 200 knots, calibrated airspeed. Those data, obtained in steady turning flight, are shown in Figure 24. Two characteristics are significant in these data. First, the cyclic control force per g remains positive throughout the range of speeds investigated although it becomes progressively lighter as g's are increased. Second, the increased control sensitivity with airspeed which was discussed in previous sections is evident here as well. Desensitizing the longitudinal control as a function of airspeed, as suggested above, would produce similar benefits in these steady-state maneuvers. It would also permit a reduction in control forces at the low speeds where the pilot felt that the maneuvering stability was too strong.

In dynamic pull-up maneuvers, the maneuvering stability remained positive but appeared to be reduced in magnitude. This is characteristic of all aircraft and is considered satisfactory.

Autorotation Entries

Autorotation entries were investigated progressively over the flight envelope up to and including speeds of 195 knots calibrated airspeed. At all speeds, the pilot reported no difficulty in maintaining proper control of rotor rpm. Control effectiveness remained strong, and

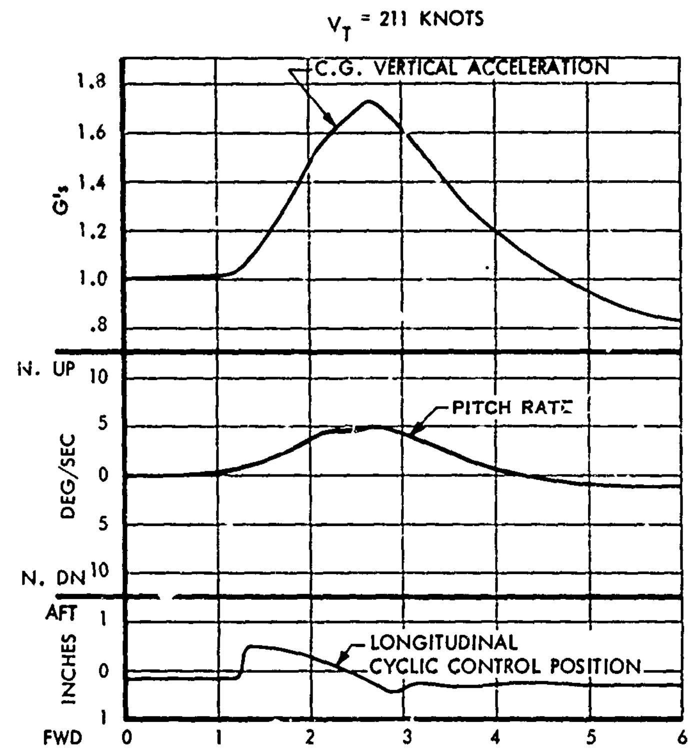


Figure 20. Time History of Longitudinal Control Power - 211 Knots

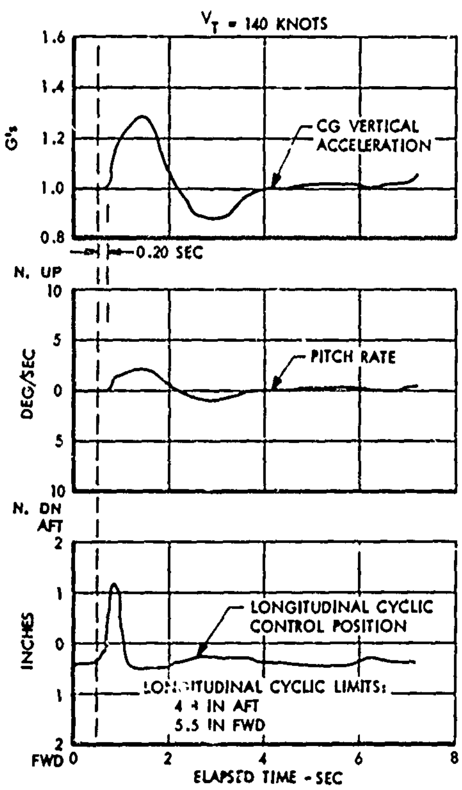


Figure 21. Longitudinal Damping and Control Response Lag - 140 Knots

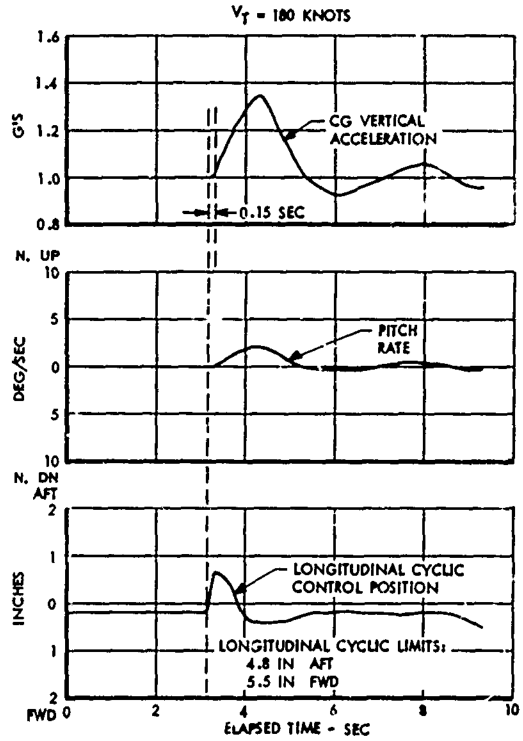


Figure 22. Longitudinal Damping and Control Response Lag - 180 Knots

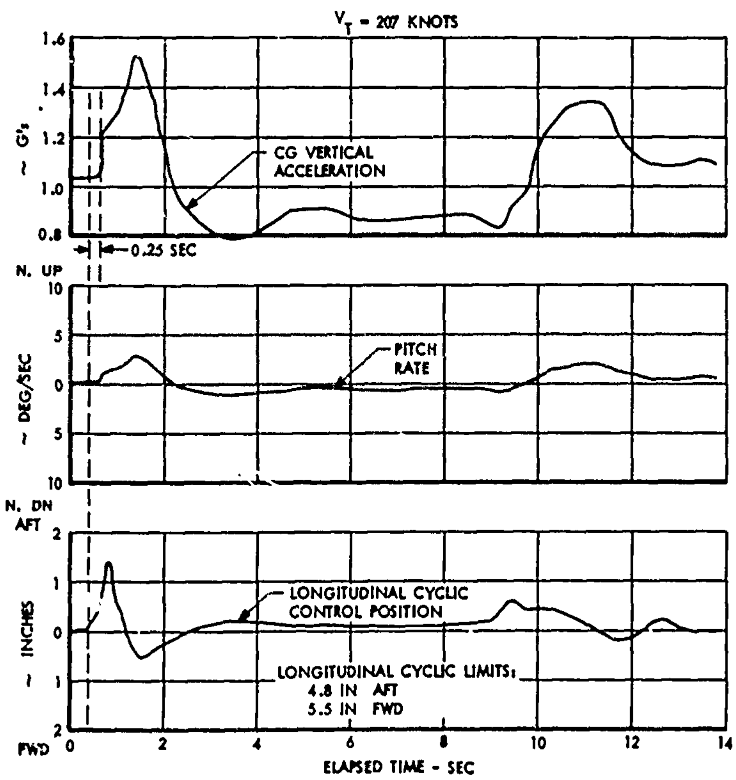


Figure 23. Longitudinal Damping and Control Response Lag - 207 Knots

maneuvering was performed with ease.

Although the rigid rotor, in operating at near zero lift, has an advantage in that its drag is low and hence rate of rpm decay is reduced, it still requires new autorotation entry techniques at high speed. The simplest method found during these tests was for the pilot to deploy the spoilers as soon as the engine failure was sensed (or simulated). This permitted the relatively high rotor angles of attack required for autorotation to be developed without producing excessive loads on the aircraft. Once the spoilers were deployed, a climbing turn was entered to produce the necessary increase in angle of attack and to assist in decelerating the aircraft to more conventional speeds. During this maneuver, the pilot could easily control rotor rpm by simply increasing or decreasing his rate of turn. Auxiliary engine thrust was reduced to idle soon after the autorotation was entered to further assist the deceleration to lower flight speeds. Once these speeds were attained, the auxiliary thrust was modulated to continue flight and permit an optimum landing site to be selected. During tests such as these it was shown that level flight could easily be maintained and the helicopter could be operated quite satisfactorily as an autogyro.

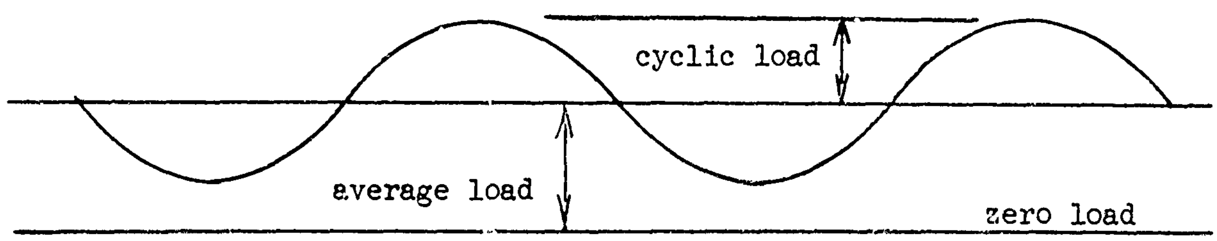
A time history of an autorotation entry at the maximum speed demonstrated for such a maneuver is shown in Figure 25. These data indicate that all characteristics are normal in all respects.

STRUCTURAL LOADS

General

Structural loads were measured during all phases of the speed and maneuvering envelope expansion of the compound helicopter. Measurements that were included during the testing are the main rotor hub and blade loads, main rotor pitch link axial loads, control gyro arm loads, wing bending, main rotor lift, horizontal stabilizer loads, and tail rotor loads. These structural loads were recorded and examined prior to each additional envelope expansion to determine the magnitude of loads and maintain safety of flight.

In this report, the load measurements are divided into two components; cyclic load and average load. The sketch below indicates what these components mean.



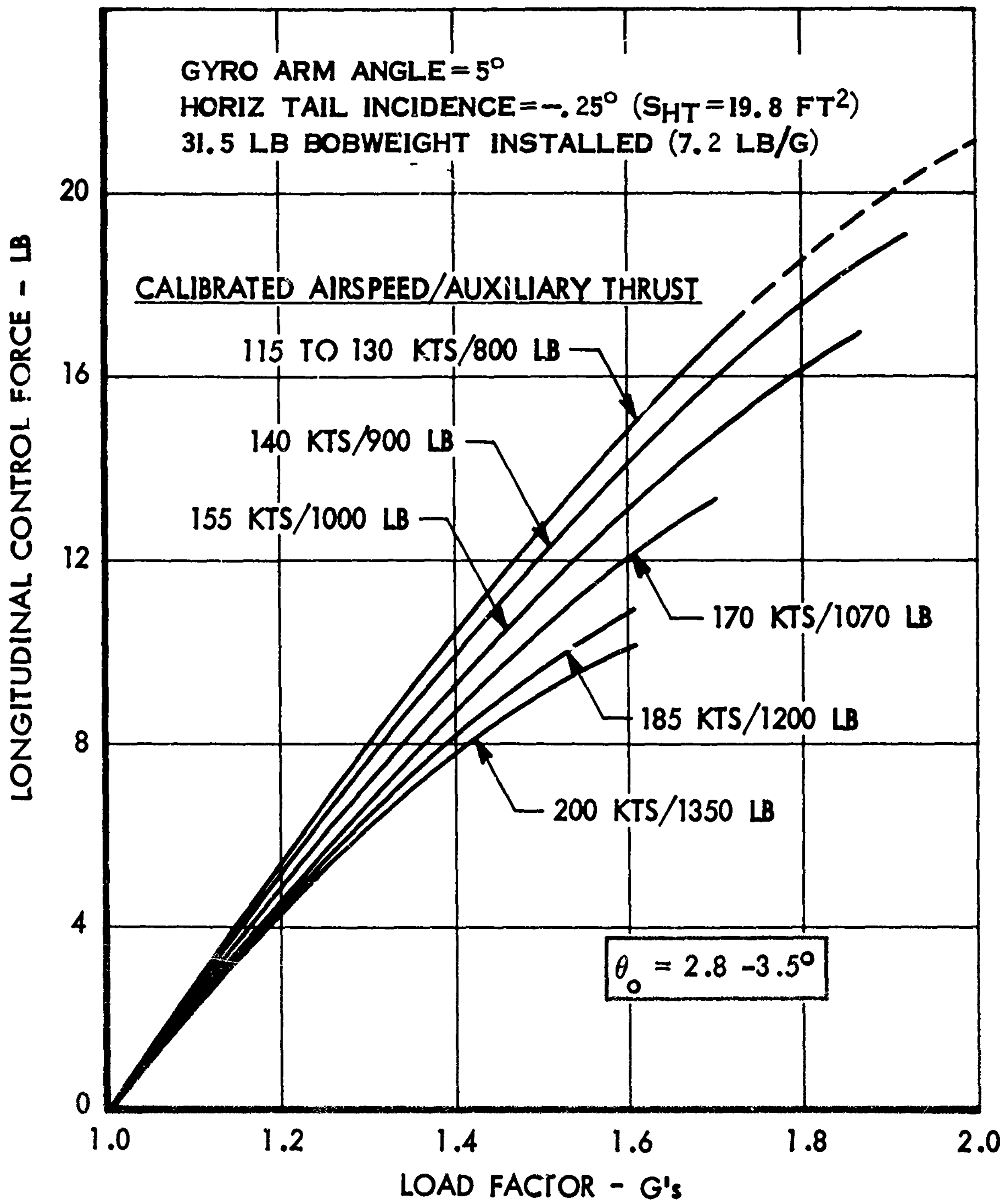


Figure 24. Maneuvering Stability

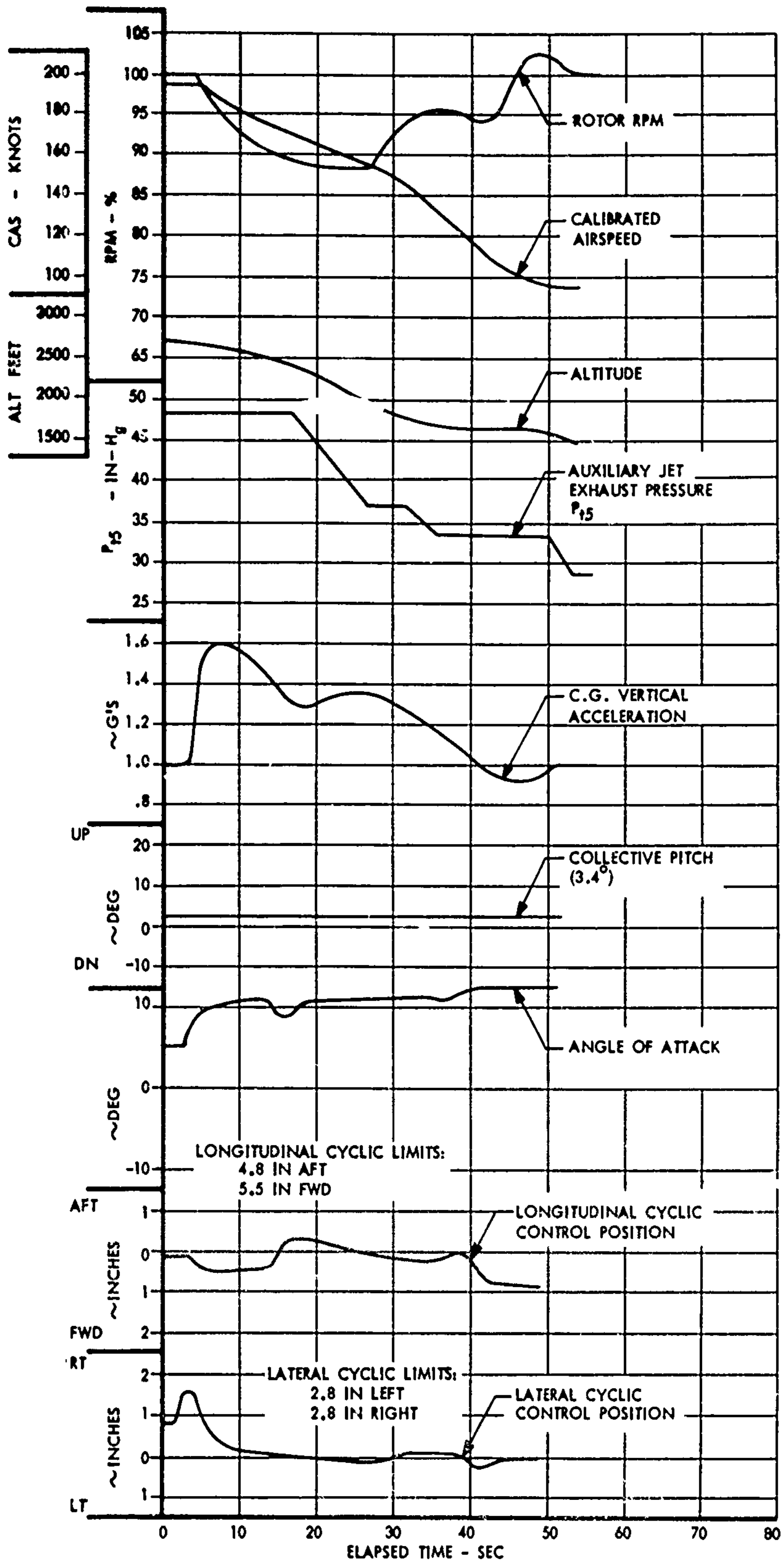


Figure 25. Time History of Autorotation Entry

A review of all structural data indicates that, for the four-way hub, station 7.0 is the critical fatigue section of the main rotor. Assuming a stress concentration factor of 3, the estimated endurance limit stress is 26,000 psi. The strain gage calibration was effected in terms of bending moment rather than stress because the bending moment curve along the span of hub and blade is predictable. The conversion of bending moment to stress at station 7.0 is as follows:

Flapwise bending moment at station 6.0 x 1.42 = stress at station 7.0
Chordwise bending moment at station 6.0 x .152 = stress at station 7.0

Level Flight

Main Rotor Loads. Tests were conducted to determine the optimum incidence angle of the horizontal stabilizer with the reduced area.

The incidence angle was initially set at 1.50 degree leading edge down from the fuselage reference line, and tests were conducted with various collective blade angle settings at various speeds. Harmonic analyses were run on the wave form of the flapwise bending moment at station 6 to determine the one-per-revolution components. This component was then resolved into a roll component and pitch component and plotted versus calibrated airspeed as shown in Figure 26. This plot shows that the main rotor blades were bending up a sizable amount in the aft position due to an excessive down load on the horizontal stabilizer. To alleviate this condition, the incidence angle of the horizontal stabilizer was changed from 1.50 degree leading edge down to 0.75 degree leading edge up. Tests were then conducted with this setting and also plotted as shown in Figure 26. This curve shows the incidence angle was now too much leading edge up. Therefore, the incidence angle was changed to 0.25 degree leading edge down from the fuselage reference line. Data from this series of tests are also shown in Figure 26. The setting of 0.25 degree leading edge down was used for the rest of the testing including the high-speed points. This setting was used because it kept the pitching moment component at the maximum speed condition as close to zero as possible, thus keeping the one-per-revolution blade flapwise bending loads as low as possible.

The main rotor blade overall flapwise cyclic bending at station 6, shown in Figure 27 for the low altitude condition, increased with speed to a maximum value of 15,000 inch-pounds at 220 knots calibrated airspeed. The maximum flapwise cyclic bending at 10,000 feet was 14,000 inch-pounds at 191 knots calibrated airspeed.

The cyclic flapwise bending at station 6 of 15,000 inch-pounds converts to a cyclic stress of 21,300 psi at station 7. The cyclic chord-

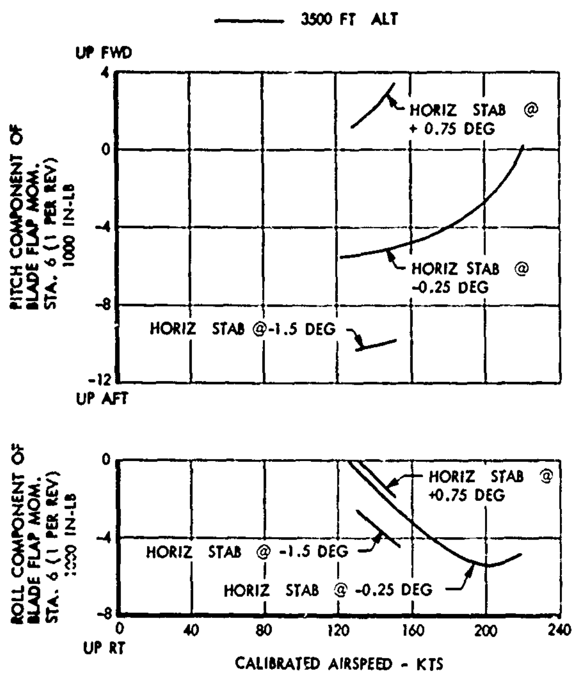


Figure 26. Roll and Pitch Component Versus Calibrated Airspeed

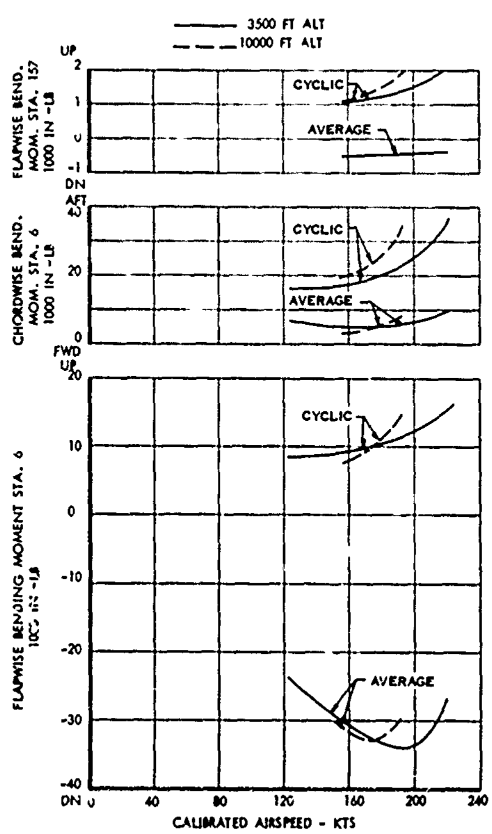


Figure 27. Main Rotor Blade Loads Versus Calibrated Airspeed

wise moment at station 6, as shown in Figure 27, was 36,600 inch-pounds; this converts to a stress of 5,500 psi at station 7. The sum of the two results in a maximum possible cyclic stress at the corner of the hub at station 7 of 26,800 psi as compared to an estimated endurance limit of 26,000 psi. These flapwise and chordwise loads were also plotted versus true airspeed in Figure 28. This curve shows that the loads are more a function of true airspeed than they are of calibrated airspeed.

The flapwise bending at station 157, recorded as a representative station of the main rotor blade outer section, is shown in Figure 27. The maximum cyclic load at this station was 2,000 inch-pounds which is well below the endurance limit.

The lift on the main rotor was recorded to determine the amount of lift that was carried by the main rotor blades as compared to the wing and fuselage lift. This rotor lift is plotted versus calibrated airspeed in Figure 29.

At the maximum speed obtained, 220 knots calibrated airspeed, the overall flapwise and chordwise cyclic loads did not rise as rapidly as the cabin vibration. From a visual observation of the wave form of the flapwise bending moment, it was determined that the three- and five-per-revolution components were rising considerably faster than the overall load. This was true for both the 3,500 foot and 10,000 foot tests. Harmonic analyses were run on the wave form of the flapwise bending moment at station 6 to determine the magnitude of the various components. The various frequencies were then plotted versus calibrated airspeed, as shown in Figure 30. The predominate frequency of three per revolution was plotted versus true airspeed in Figure 31. Also, the various frequencies were plotted versus the advancing main rotor blade tip Mach number, Figure 32. From these curves it was determined that the sharp rise in the three- and five-per-revolution components was a function of tip Mach number rather than calibrated or true airspeed. The maximum Mach number of the advancing blade tip was 0.925 which is well above the critical Mach number for a 12% thick blade section as described in the Flying Qualities section of this report.

The overall chordwise bending moment at station 6 and the pitch link axial load were plotted versus the advancing blade tip Mach number as shown in Figure 33. This plot shows that these parameters are a function of tip Mach number, but they do not rise sharply at Mach 0.90 as the flapwise loads did.

Pitch Link Loads. The maximum cyclic pitch link axial load was only 315 pounds, as compared to an estimated endurance limit of 1,400 pounds as shown in Figure 29. The pitch link load increased linearly with speed until the tip speed of the advancing main rotor blade was approximately Mach 0.90. At this point, there was a rise in cyclic loads as shown

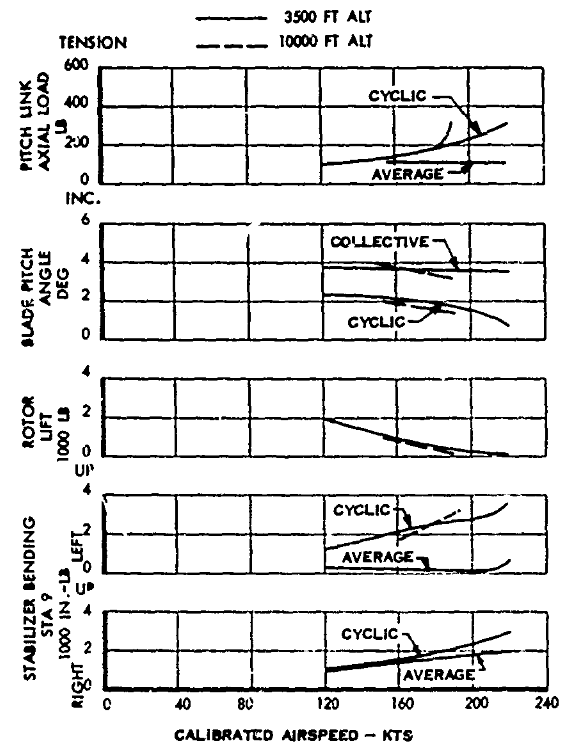


Figure 28. Main Rotor Loads Versus True Airspeed

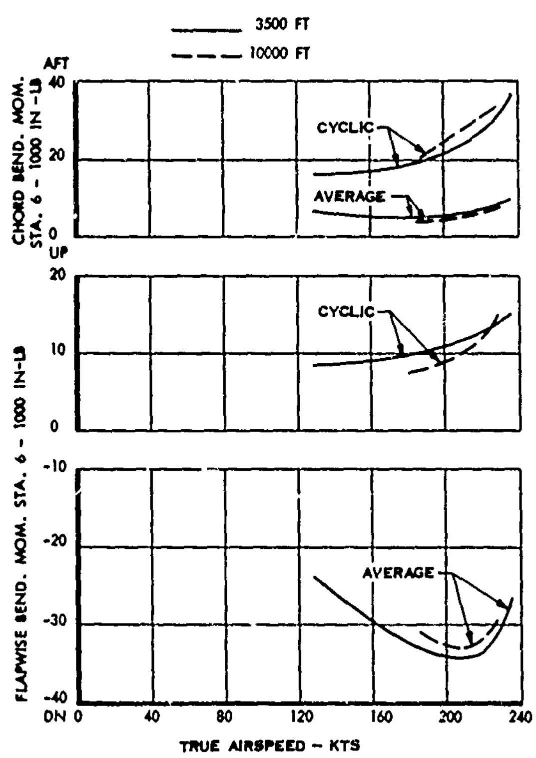


Figure 29. Rotor Characteristics Versus Calibrated Airspeed

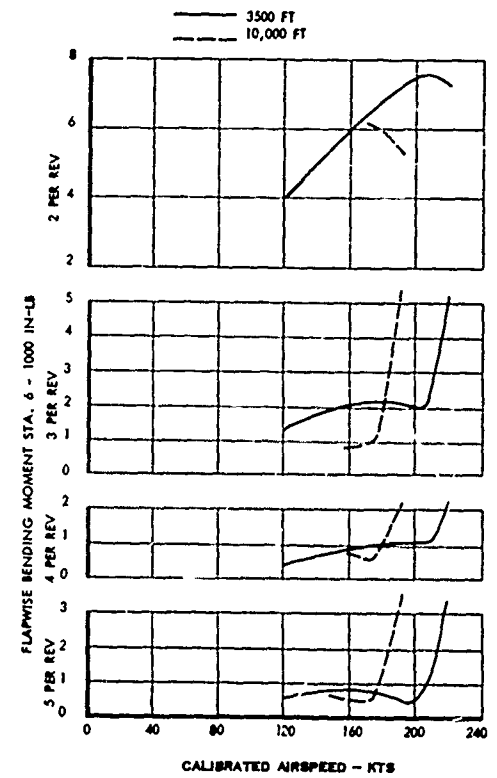


Figure 30. Harmonic Components of Flapwise Bending Versus Calibrated Airspeed

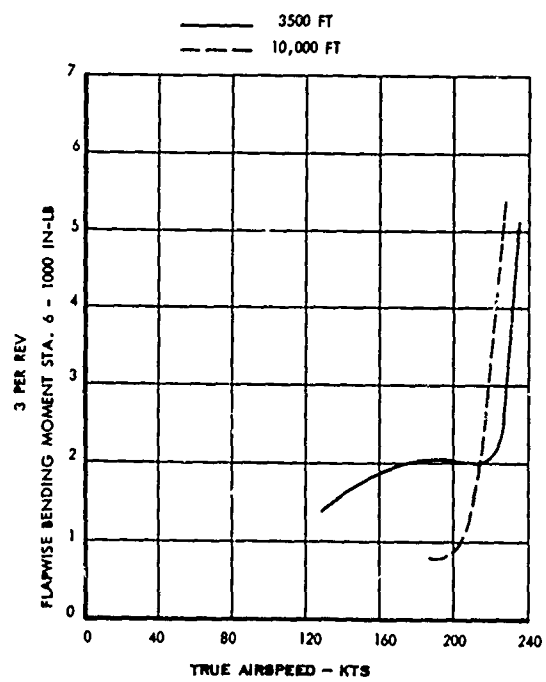


Figure 31. Flapwise Bending Moment Versus True Airspeed

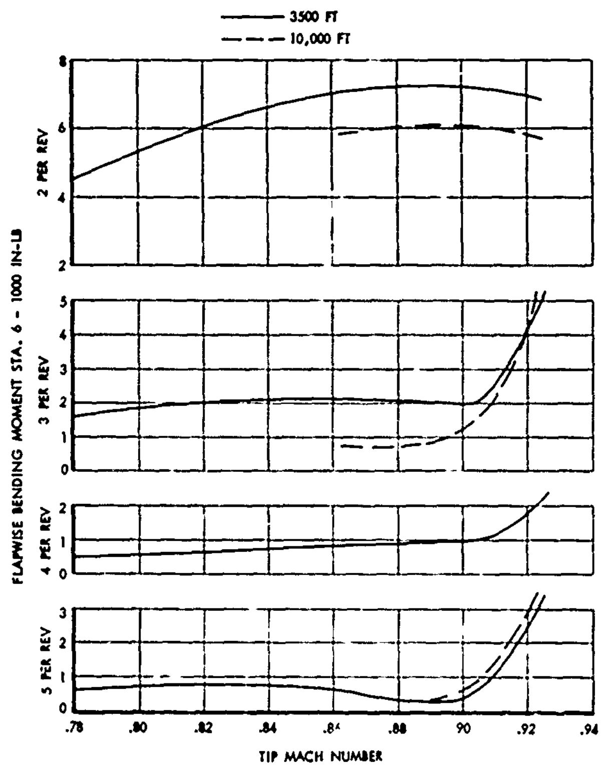


Figure 32. Harmonic Components of Flapwise Bending Versus Tip Mach Number

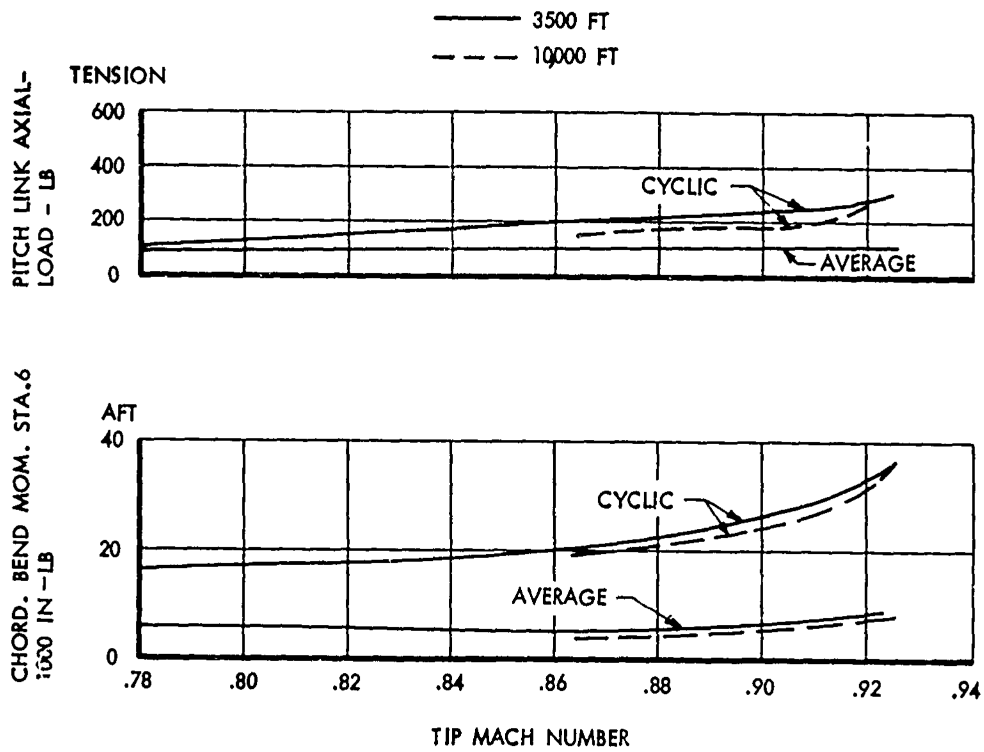


Figure 33. Rotor Loads Versus Tip Mach Number

in Figure 33. The rise was apparently due to the Mach number effect on the flapwise and chordwise bending of the blade.

Collective Pitch. Tests were conducted with high collective pitch, approximately 6.3 degrees, and then with low collective pitch, approximately 2.4 degrees. These tests were conducted over a speed range of 92 to 188 knots calibrated airspeed to determine the effect of different collective pitch settings. The more important measurements that show the variations as a result of these collective pitch settings are plotted in Figures 34 and 35. These plots show that the high collective pitch results in an increase of approximately 1,000 pounds in rotor lift over the low collective pitch setting. This increase in lift resulted in an increase in cyclic blade pitch angle, cyclic pitch link axial load, and chordwise loads at station 6. However, the blade flapwise loads at station 6 were reduced at the higher collective pitch setting.

Horizontal Stabilizer Loads

From the previous tests of the compound helicopter it was determined that the cyclic structural loads of the horizontal stabilizer were above the estimated structural endurance limit of 3,200 inch-pounds. To alleviate these loads, the area of the horizontal stabilizer was reduced from 24.2 to 19.8 square feet by cutting 12 inches off the tip of each horizontal stabilizer. This reduced area raised the symmetrical first bending mode of the stabilizer from 30.5 cps to 42 cps, which is above the tail rotor rotational frequency of 35 cps. This reduced area also raised the antisymmetric bending frequency from 20.5 cps to 22.9 cps. During ground shake tests of the smaller horizontal stabilizer, an attempt was made to stiffen it by adding struts from about the 2/3 span point on each side of the horizontal stabilizer to the bottom of the fuselage. However, the shake tests showed no effect of the struts on the symmetric or antisymmetric frequencies, so the strut was omitted for the first flight.

Initial flight tests with the new stabilizer showed that the overall cyclic loads were approximately the same magnitude as had been obtained with the larger tail. However, the mode had changed from a symmetrical mode at the tail rotor rotational frequency of 35 cps to an unsymmetrical bending natural frequency of 22.9 cps to the aft fuselage rolling frequency of 24 cps due to four per revolution from the main rotor caused the shift in stabilizer mode. The ability of the stabilizer attachment structure to withstand unsymmetrical moments is considerably less than its ability to take symmetrical loads, thus further changes in stabilizer configuration were required. As a first change, the struts were reinstalled to see if they might be more effective in flight than they were on the ground shake tests. Test results showed they did help

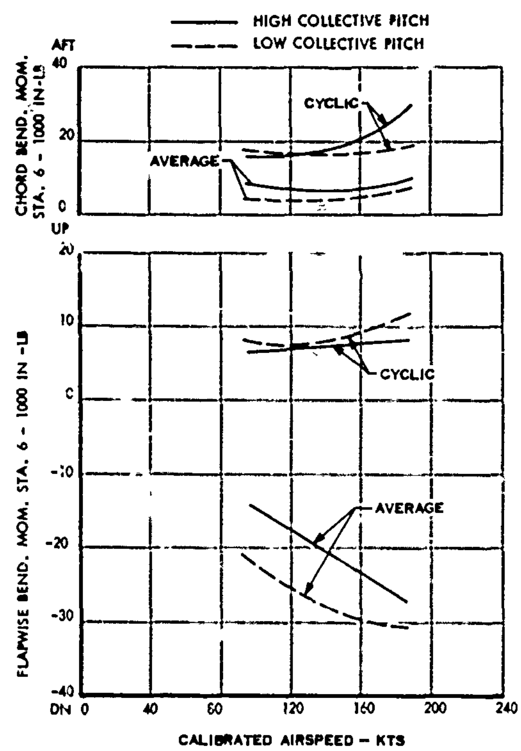


Figure 34. Main Rotor Loads Versus Calibrated Airspeed

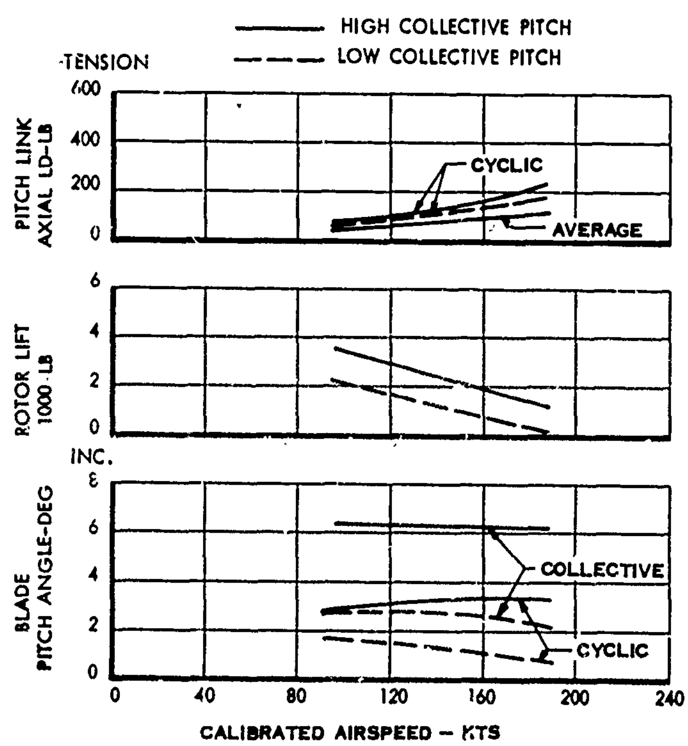


Figure 35. Rotor Loads Versus Calibrated Airspeed

reduce the loads a reasonable amount, but the loads were still higher than desired. Therefore, it was decided to add weights to the tip of each stabilizer to reduce both the symmetric and antisymmetric frequencies. After some preliminary ground tests to determine the amount of weight required, 8-pound weights were added to each stabilizer tip with the struts installed. The symmetric bending frequency of this configuration was determined to be 29.8 cps. The actual antisymmetric frequency was not determined, but it was considered sure to be below the original 20.5 cps of the larger horizontal stabilizer.

Flight tests of this configuration showed that the basic bending mode of the stabilizer had returned to the symmetric mode at the tail rotor rotation frequency (35 cps) and the overall loads were somewhat lower than obtained with the original large tail. Although the basic bending mode was symmetric, the right and left loads were not equal and varied in amplitude at different phase relationships. This produced an unsymmetric component of stabilizer load that was still higher than desirable from a fatigue standpoint.

A close review of the horizontal stabilizer attachment strength revealed that increasing the size of some of the bolts holding the stabilizer attachment brackets from 3/16 to 1/4 inch would provide a sufficient increase in strength to eliminate the unsymmetrical load problem. Also, the basic bending strength of the stabilizer was reviewed, and it was decided that the symmetrical fatigue limit could be safely raised from 3,200 inch-pounds to 4,000 inch-pounds (4,000 inch-pounds is roughly 10% of the calculated ultimate bending strength). The horizontal stabilizer loads are plotted versus calibrated airspeed in Figure 29. These loads are somewhat below the estimated structural endurance limit at all speeds up to and including the maximum speed tested.

Tail Rotor Loads. Further analysis of the tail rotor loads from the previous testing indicated that the loads were higher than those previously reported in reference 2 due to a galvanometer's response that was down approximately 30% in the 104-cps area. Also, analysis of the tail rotor loads showed that the major load was at a frequency of three per revolution. These loads are produced by the first antisymmetric elastic mode (flap bending opposite of teetering motion) of the tail rotor which has a natural frequency near three per revolution.

This analysis indicated that the peak moment occurs around station 12 (cuff area) and that the highest stresses occur at station 16.8 where the doublers end on the basic blade section. The moment ratio between station 16.8 and the station where the strain gages are located was 1.21. Therefore, to correct for this, the estimated endurance limit of 790 inch-pounds for the basic blade section was reduced to 660 inch-pounds.

During the initial testing the tail rotor loads were higher than desired with the revised structural endurance limit. It was concluded that aerodynamically unloading the tail rotor would help alleviate this problem. Accordingly, the trailing edge vertical fin tab was bent 10 degrees to the left. Measurements obtained with this configuration are plotted versus airspeed in Figure 36. The tab bend helped some but not enough since the endurance limit was exceeded at about 180 knots.

To further alleviate the tail rotor loads, the fin tab bend was increased to 20 degrees to the left. Loads measured with this configuration are also plotted versus airspeed in Figure 36. With this tab increment the endurance limit was exceeded at about 200 knots. To provide even greater relief to the tail rotor loads, the area of the fin tab was increased by 30%. Loads measured with this increased area tab were below the endurance limit at the maximum speed tested as shown in Figure 36.

The only condition that remained critical on the tail rotor after the tab was adjusted to take care of the high-speed case was during transition after takeoff using high jet thrust. This high jet thrust produced a yawing moment that had to be counteracted almost entirely by the tail rotor due to the ineffectiveness of the vertical fin at these low speeds. This high loading condition was alleviated by reducing the thrust of auxiliary jet until after the helicopter had accelerated to higher speeds.

Maneuvering Conditions

The load factors obtained at various true airspeeds up to 236 knots are shown in the maneuvering envelope of Figure 37. The maximum load factor was 2.12g at 133 and 172 knots true airspeed and the minimum load factor was .27g at 156 knots true airspeed. The same load factors are plotted versus calibrated airspeed in Figure 38. The above load factors are corrected to 4,500-pound gross weight.

The main rotor flapwise bending moment at station 6 is plotted versus load factor, Figure 39. This plot shows the average load increasing with load factor due to the increasing lift on the main rotor blades. The scatter in average loads at the higher load factor is due to the fact that these tests were run at various speeds where the percentage of lift carried by the wing changes rapidly with speed, thus relieving part of the load that the main rotor blade must lift to produce the load factor. The maximum cyclic flapwise bending moment was 25,500 inch-pounds.

Main rotor chordwise bending moment at station 6 is plotted versus load factor, Figure 40. This plot shows that the average loads are not a function of load factor. The cyclic loads are affected some by load

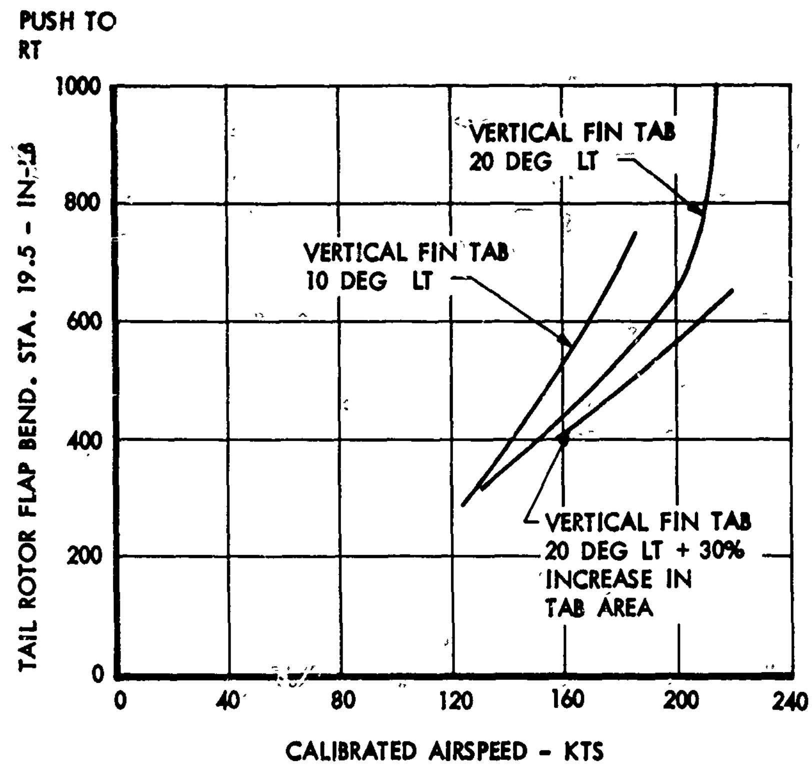


Figure 36. Tail Rotor Flapwise Bending Versus Calibrated Airspeed

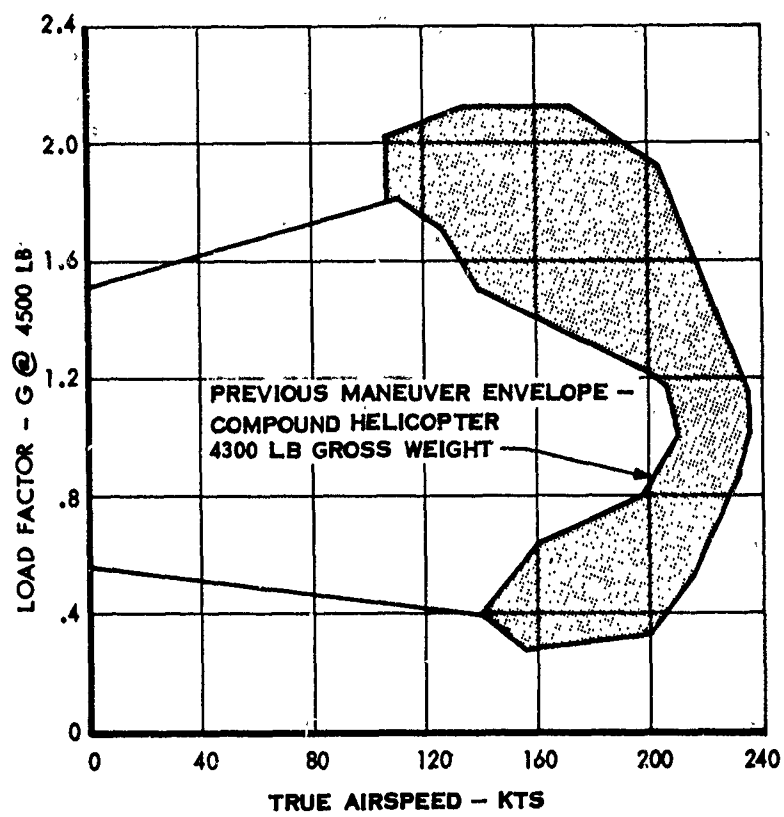


Figure 37. Maneuvering Envelope Versus True Airspeed

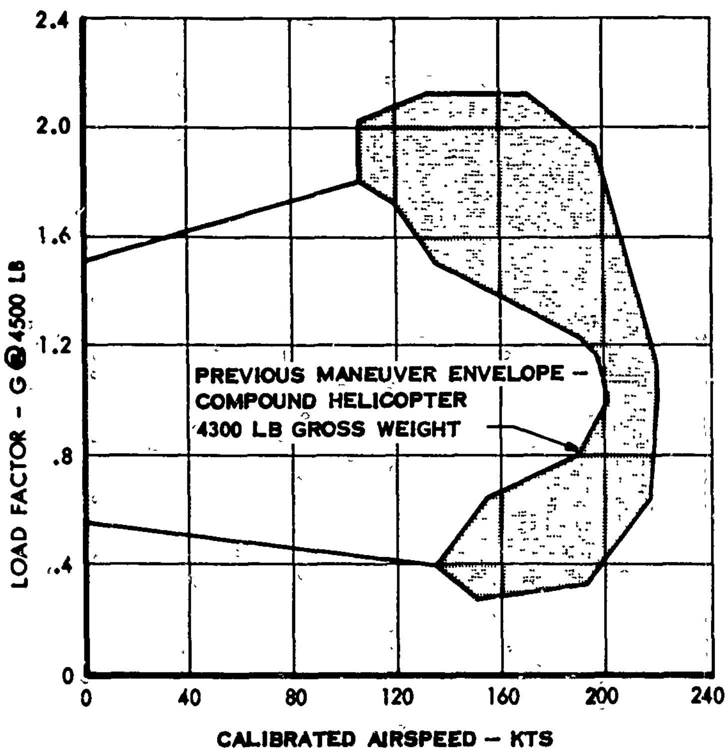


Figure 38. Maneuvering Envelope Versus Calibrated Airspeed

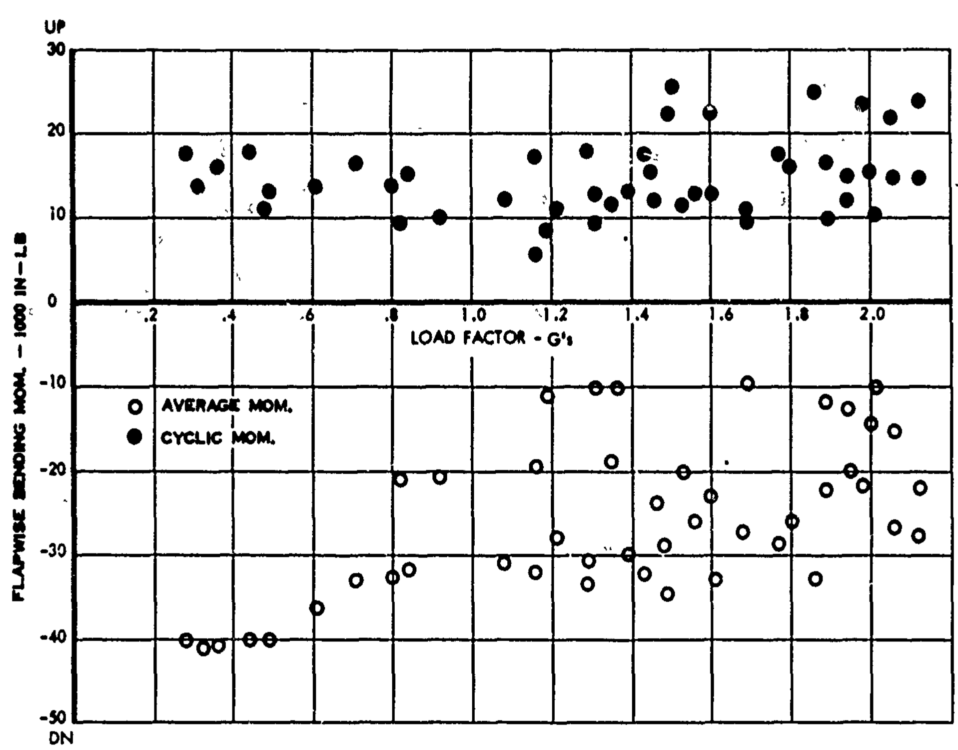


Figure 39. Main Rotor Flapwise Bending Moment Versus Load Factor

factor with the highest values occurring at the higher load factors. The maximum cyclic load was 43,000 inch-pounds.

The flapwise and chordwise cyclic loads at station 6 that are plotted are the maximum loads that occurred during the maneuver and do not necessarily occur at the time of the maximum load factor. The flapwise loads are more a function of the rate of the pilot's input than the amount of load factor that is produced during the maneuver. The chordwise loads are affected by both pilot input and load factor.

Autorotation

Structural loads were measured during the transition from powered flight to autorotation and during the autorotation for various speeds up to and including the maximum autorotation entry speed of 195 knots calibrated airspeed. The flapwise bending loads increased slightly during the transition from powered flight to autorotation and then decreased well below the normal powered flight loads. The other structural parameters were usually less during autorotation entry and during autorotation than those experienced in powered flight.

VIBRATION

Vibration level in the cabin was measured for speeds up to and including the maximum speed of 220 knots calibrated airspeed at 3,500 feet, and to 191 knots at 10,000 feet as shown in Figure 41. This plot shows a high rise in vibration level when the advancing tip of the main rotor blade approached Mach 0.90 due to the rise in three- and five-per-revolution cyclic moments of the main rotor as shown in Figure 32. The cabin vibration was also plotted versus the three-per-revolution flapwise bending moment at station 6 to indicate the relationship between these two parameters as shown in Figure 42. There was a reduction in vibration at the higher altitude for a given three-per-revolution flapwise input as shown in the curve.

PILOT OBSERVATIONS

The following comments have been submitted by the pilot who conducted the test flights.

During this program, the speed and maneuvering objectives were completed and exceeded to a good degree. Additional flights were conducted in a limited scope to determine the aircraft longitudinal and lateral characteristics. The major portion of the flights were conducted at or below a 3,500-foot pressure altitude with some high "g" flights conducted at sea level at speeds up to 210 knots calibrated airspeed. In addition, three flights at or above a 10,000-foot pressure

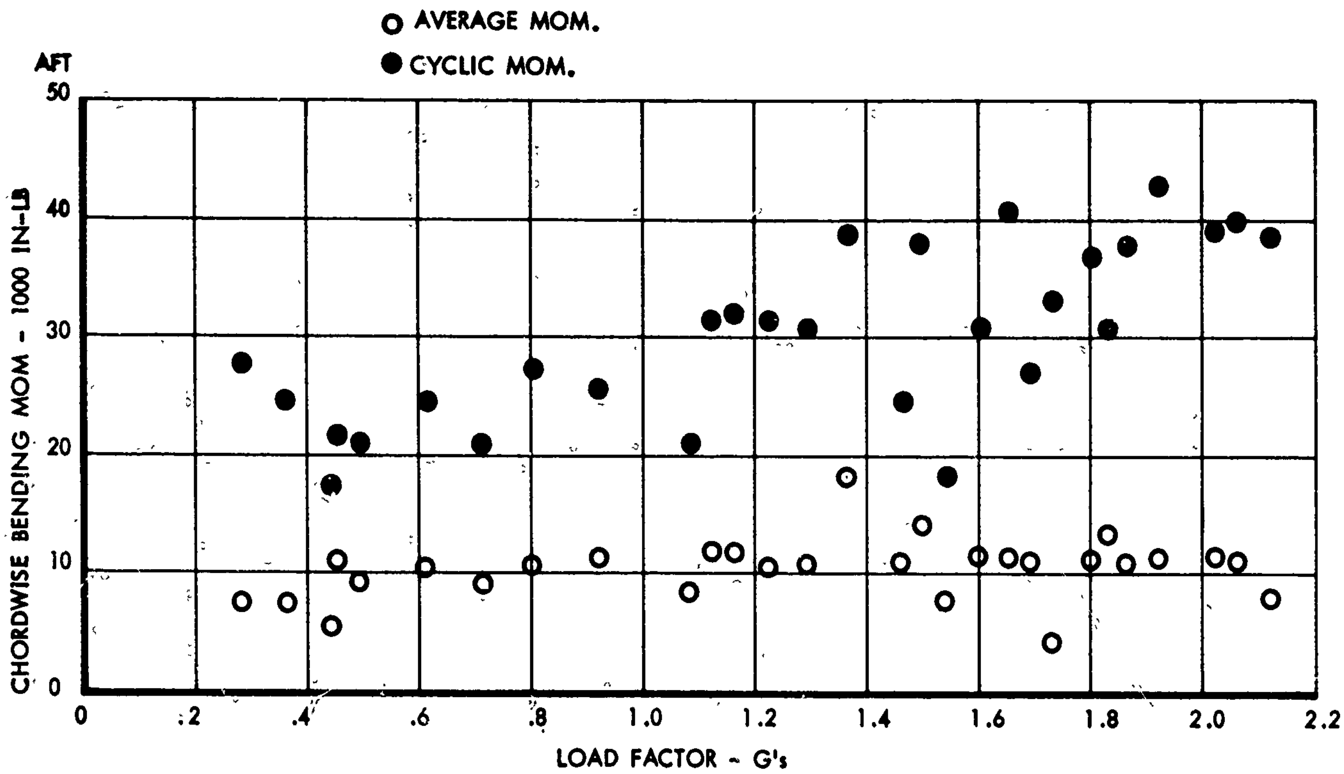


Figure 40. Main Rotor Chordwise Bending Moment Versus Load Factor

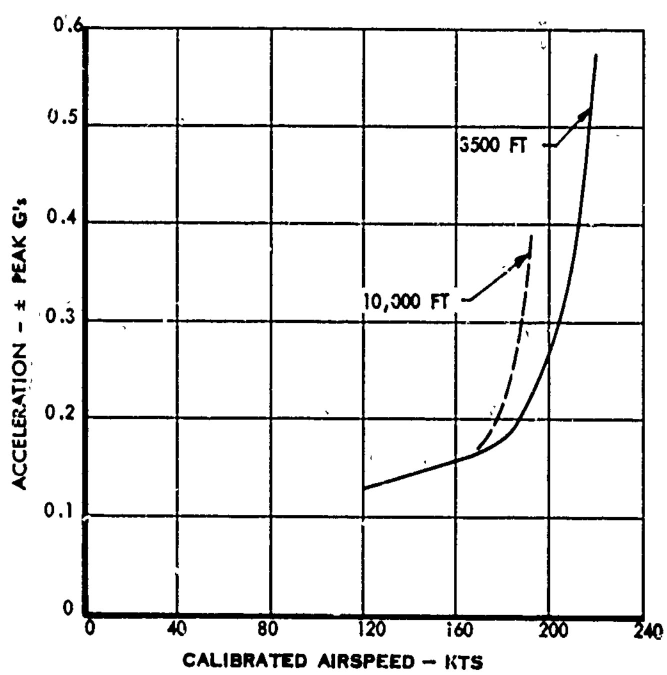


Figure 41. Cabin Vertical Vibration Versus Calibrated Airspeed

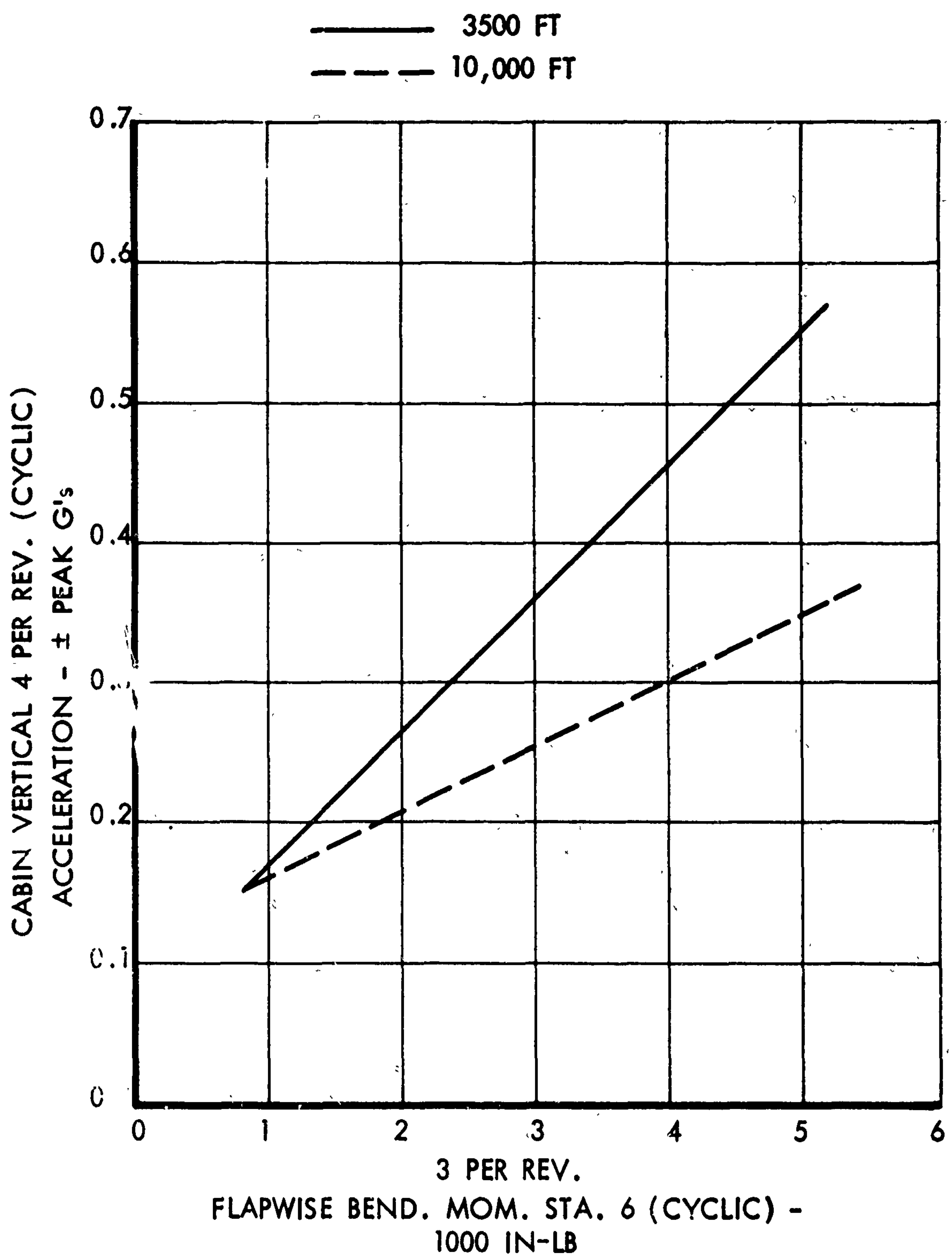


Figure 42. Cabin Vertical Vibration Versus Flapwise Bending Moment

altitude were conducted with density altitudes in the vicinity of 12,000 feet. At the conclusion of the program, several operational or "tactical" maneuvering flights were scheduled for the pilot to evaluate the compound aircraft handling and performance qualities as "he saw fit". These flights added immeasurably to the pilot's knowledge with respect to the operational characteristics of this configuration as compared with a pure helicopter or airplane.

Stability and Control

Qualitative and quantitative stability and control flights have not revealed any new or unusual characteristics with respect to longitudinal dynamic stability as reported within the speeds evaluated during the previous program.

Above approximately 210 knots the dynamic longitudinal short period was well damped. However, the period appeared to change slightly with speed, and the phasing to control inputs began to induce some over-controlling as was the case below 140 knots.

The phugoid was damped and the period was in excess of 20 seconds. In these tests the complete cycle could not be completed because of the low ceiling existing at the time of the tests. The phugoid was also influenced by longitudinal control friction. This was more apparent with increased speed where the control effectiveness had increased some. Control friction would influence the period to make it appear to be divergent nose-up or nose-down depending on the ability to return the stick to neutral and the fineness of trim by the pilot. In these instances, the low ceiling would not permit airspace for the aircraft to return to trim or diverge to an unusual attitude. When the control was manually returned to trim, the phugoid did return toward trim and up to 3/4 of a cycle was performed under the low ceilings in some cases.

With respect to a short period induced by a gust disturbance at or near V_{max}, the short period would at times be followed by a nose-down phugoid. The first impression was that of a gentle tuck tendency until additional qualitative investigation revealed that the friction band and/or a slightly nose-down out-of-trim was the cause. It was also noticed that the bobweight would, during a gust disturbance, shift the control position slightly forward of the lost motion of the control system and would influence the nose-down direction of a phugoid type maneuver. This characteristic was not as noticeable below 205 knots, possibly because of the phasing of control input to the short period.

Also, during the last 8 to 10 knots of speed extension, both a longitudinal and lateral trim shift were noticed. Longitudinal control position began to shift slightly to the rear above about 210 knots calibrated airspeed. The shift was small with respect to motion but was more apparent as a stick force change.

Longitudinal control motion characteristics did not appear to change appreciably with variation of the collective position. The longitudinal control shift did not appear to influence overall handling characteristics except perhaps as they may have affected out-of-trim and the dynamic characteristics (discussed above) at or near maximum speed.

One of the important factors which limits maximum speed is control margin remaining. In this respect, lateral control requirements influenced the speeds obtained. At approximately the same speed at which the longitudinal control motion shift was first apparent, the lateral stick trim position also shifted quite strongly. Left rolling moments seemed to require noticeable right lateral trim to remain wings-level. At V_{max} approximately one-third lateral control remained. Although roll rate to the right was less than optimum for tactical use, more than ample control power was remaining at this setting to handle any unusual situation that could be expected to occur. Lateral trim shift was also influenced by collective position. Previously, a higher collective setting of 3.2 (pilot's indicator units)* was used with only minor variations. Investigation revealed that with a lower collective setting of 2.8 units*, lateral trim shift would be reduced. This lower setting was also found to be adequate for autorotation and maneuvering flight.

Steady-state maneuvering flight at constant power, auxiliary thrust and speed (stick force per g) indicate that the optimum longitudinal maneuvering characteristics are in the speed range of 175 knots to 195 knots. Stick force per constant g appears to be too high below 155 knots down to 100 knots in maneuvers of 2.0 to 2.5g (observed) when held for 180 to 360 degrees of turn. In all cases, stick force per g is positive whether in a steady maneuver or a tactical pull-up-type maneuver. At 195 knots and above, it was felt the Fs/g was increasing strongly with g up to approximately 1.8 observed g. Here, it was felt that the stick force remained more or less constant with increased g. These observations are qualitative and preliminary only and should be investigated more thoroughly before conclusions can be made, however.

Lateral-directional stability characteristics remain essentially as discussed in reference 2 over the entire speed range.

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* Pilot's collective position indication of 3.2 units corresponds to 3.8 degrees θ_0, 2.8 units is 2.9 degrees θ_0.

In the pure phugoid maneuver a right roll is introduced with nose-up attitude and conversely. During rate of roll investigation, the roll rate to the left seemed greater than to the right by a noticeable amount. Left roll rate appeared to be less than that of the pure XH-51A per inch of stick displacement and more on the order of a T-28 or OV-1 for the first inch of stick displacement. The opinion of this pilot is that this is less than optimum for an operational aircraft of this type. Right roll rate seemed slightly greater than that of conventional small helicopters and although adequate should be increased.

Autorotation

Autorotations and descents were conducted up to 195 knots calibrated airspeed. Entries were conducted from a fixed collective position for level flight. Delays in spoiler action or J-60 thrust reduction did not affect the characteristics significantly. Decay to 90% rotor rpm was permitted and held at 190 and 195 knots without any noticeable influence on aircraft characteristics.

Rotor rpm was permitted to build up in steep turns or climbing turns to 110% (maximum continuous) with no difficulty and could be controlled easily with cyclic or collective as desired.

Performance

Level flight performance flights were conducted with various fixed collective positions. Climbs to altitude were made at constant J-60 throttle setting and constant exhaust gas temperature for maximum continuous power. Continuous rates of climb in excess of 3,500 feet/minute using constant exhaust gas temperature were held at 115 to 125 knots calibrated airspeed and a collective setting of 3.5 units. On all occasions, climbs and climb-turns to 10,000 feet from takeoff were conducted in under 5 minutes.

Altitude Flights

Flights at altitude did not reveal any unusual or adverse stability and control characteristics. A maximum speed of 190 knots was attained with characteristics similar to those found at sea level. At 170 knots and below, the characteristics were considered to be outstanding. The normal "helicopter feeling" with altitude flight was not present.

Vibration Levels

Vibration levels were satisfactory up to 210 knots calibrated airspeed at sea level and 185 knots calibrated airspeed at altitude. 1P vibratior was reduced some with tracking. Above these speeds a high-frequency 4P vibration (or less) is encountered. At the maximum speeds, activity in the windshield from the high-frequency vibration was the major limiting factor for further speed increases.

Noise Level

Investigation of the random noise level was conducted, and from it a negative pressure area in the cockpit and cabin was discovered. Inward deflection of the cabin rear door starting at approximately 175 knots would cause the random noise. Bracing and stiffening of the door eliminated the noise entirely.

PROBLEMS ENCOUNTERED AND SOLUTIONS

Listed below are the problems which were encountered during this program, together with a brief discussion of the solutions.

- Vibration. Probably the most significant problem area encountered was a generally high vibration which occurred above approximately 210 knots. Since the objectives of the program were exceeded, the solution to this problem was considered beyond the scope of this contract; however, possible solutions have been suggested in the Recommendations section of this report.

- Horizontal Stabilizer Loads. Cyclic loads in the horizontal stabilizer at high speed were found to be higher than desired. Further, an unsymmetrical mode was evident which produced high vibratory loads in the stabilizer attachment fittings. Both problems were solved by properly tailoring the symmetric and unsymmetric frequencies through the addition of tip weights and external struts.

- Tail Rotor Loads. Tail rotor stresses above the endurance limit were encountered at flight speeds above 180 knots. Since these high loads appeared to be associated with blade flapping, it was concluded that they could be substantially reduced by aerodynamically unloading the tail rotor. This was accomplished by adding approximately 30% to the trim tab area and by bending the tab to increase the effective camber of the vertical stabilizer. No further tail rotor problems were encountered.

- Windshield Loads. High dynamic pressure at high speed in combination with a lower-than-ambient cabin pressure produced an adverse pressure gradient across the windshield which tended to buckle it. To prevent this, additional internal bracing was installed together with a ram scoop to reduce the negative pressure in the transmission and cabin compartments. These modifications eliminated the problem, and no further difficulties were experienced in this regard.

- High Pitch Noise. At speeds above 160 knots, a high-pitch, apparently metallic noise was heard which, until the source was found, prevented any further extensions in the speed envelope. After careful investigation, it was found that the aft cabin door was deflecting inward at high speed and was vibrating noticeably. This vibration, because of close tolerances between the door and frame, produced the reported metallic noise. The solution to the problem was simply to strengthen the door in the vicinity of the window and to add rubber compression strips between the aft edge of the door and the frame. No further noises of this type were noted during the remainder of the program.

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* Formerly, U. S. Army Transportation Research Command.

Unclassified

Security Classification

DOCUMENT CONTROL DATA - R&D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author)	2a. REPORT SECURITY CLASSIFICATION
Lockheed-California Company A Division of Lockheed Aircraft Corporation P. O. Box 551, Burbank, California 91503	Unclassified 2b. GROUP

3. REPORT TITLE

Extension of the High-Speed Flight Envelope of the XH-51A Compound Helicopter

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

Final Report

5. AUTHOR(S) (Last name, first name, initial)

Wyrick, Donald R.

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
November 1965	59	3

8a. CONTRACT OR GRANT NO.	9a. ORIGINATOR'S REPORT NUMBER(S)
DA 44-177-AMC-150(T) b. PROJECT NO.	USAAVLABS Technical Report 65-71
c. Task 1P121401A14311 d.	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) Lockheed Report 18917

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13. ABSTRACT

This report presents the results of a speed extension program conducted by the Lockheed-California Company on the compound version of the rigid-rotor XH-51A helicopter (under a supplement to contract DA 44-177-AMC-150(T), Modification 1). The principle objective of this program was to investigate the flight characteristics of the compound helicopter with special emphasis on the areas of flying qualities, performance, structural loads, vibration, and maneuverability in the speed range of 200 to 230 knots true airspeed. All contract objectives were met or exceeded. The maximum level flight true airspeeds demonstrated, during this program, were 236 knots at sea level and 228 knots at a density altitude of 12,000 feet.

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