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XH-59A ABC TECHNOLOGY DEMONSTRATOR ALTITUDE EXPANSION AND OPERATIONAL TESTS

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APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

The flight envelope of the XH-59A aircraft was expanded to help define the capabilities and limitations of the Advancing Blade Concept (ABC) rotor system. A maximum true airspeed of 263 knots (KTAS) was attained in a 7-degree dive at 13,000 feet, and a service ceiling of 25,500 feet density altitude was achieved at 150 KTAS. Maximum airspeed was limited by upper rotor shaft stress which slightly exceeded the fatigue endurance limit at 263 KTAS. Higher airspeed could possibly be attained with a movable or repositioned elevator that would reduce rotor pitching moments.

Attempts made to reduce rotor speed by increasing collective pitch were frustrated by corresponding increases in rotor head stresses. This precluded a systematic evaluation of potential performance gains that would be expected from rotor slowing. Test data indicated a poor lift distribution along the blade span at high speed. At the 90-degree azimuth position, a large area of negative lift was found on the outboard portions of the advancing blades. A second-generation ABC rotor designed for high speed would use less blade twist and a different inboard airfoil to reduce rotor loads and improve rotor L/D. In spite of nonoptimum rotor geometry, the XH-59A demonstrated approximately 2g capability at 225 knots and 0g between 115 and 235 KTAS.

Operational tests at Fort Rucker, Alabama, in which the XH-59A was flown by Government pilots in nap-of-the-earth and contour flight modes, confirmed the excellent handling qualities in most flight regimes and verified advantages of a compact configuration.

Results of this program will form a data base from which subsequent ABC designs can evolve. Technical areas requiring further R&D effort to exploit the potential of the ABC rotor have been identified in this report and in USAAVRADCOM TR 81-D-5.

Mr. Harvey R. Young, Aeronautical Division, was the project engineer for this program, and Mr. Duane R. Simon was the project test pilot.

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This report presents the results of two flight test programs conducted with the XH-59A Advancing Blade Concept ~~(ABC)~~ TM demonstrator aircraft. The first program, conducted in the auxiliary propulsion configuration, consisted of altitude and center-of-gravity envelope expansion testing. Test results verified the concept of developing lift primarily on the advancing rotor blades to dramatically improve lift and speed potential. The XH-59A achieved a maximum speed of 263 knots true airspeed and demonstrated a service ceiling of

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25,500 feet. The entire flight envelope was demonstrated without classical retreating blade stall. Rotor loads and stresses and rotor tip clearance followed predicted trends and were controllable. Limited center-of-gravity envelope expansion was accomplished.

The second program consisted of operational tests conducted primarily in the nap-of-the-earth and contour flight environments, using both the auxiliary and helicopter propulsion modes. The compact ABC design and smaller rotor diameter enhanced operation in confined areas. Operational test results showed the ABC to be a viable alternative concept for future tactical aircraft.

SUMMARY

The Advancing Blade Concept (ABC) demonstrator aircraft completed two follow-on test programs to expand the altitude and center-of-gravity envelopes and to test the aircraft in an operational environment. Envelope expansion tests were conducted in the auxiliary propulsion configuration. Operational tests were conducted in both the helicopter and auxiliary propulsion configurations.

Altitude envelope expansion test results have verified the concept of developing lift primarily on the advancing blades of a rotor to dramatically improve lift and speed potential. During the test phase, the XH-59A achieved a maximum speed of 263 knots true airspeed (KTAS) and demonstrated a service ceiling of 25,500 feet. A significant maneuver envelope was opened through 240 KTAS. Rotor speed reduction to 78 percent was demonstrated at lower airspeed (120 KTAS). The present rotor configuration precluded significant reductions at higher airspeeds.

Rotor loads and stresses and rotor tip clearance followed predicted trends; they were controllable throughout the flight envelope. The entire flight envelope, including nondimensional blade loadings up to $C_T/\sigma = 0.28$, was demonstrated without classical retreating blade stall.

The aircraft's flying qualities are very good in both the low- and high-speed flight regimes. Stability augmentation system (SAS) gains selected to assist the pilot are low, and the entire envelope to maximum level-flight speed and through maneuvers has been flown with the SAS off. The high control power available has resulted in a very maneuverable aircraft, and high damping precludes any tendency to overcontrol.

Limited center-of-gravity envelope expansion was accomplished at 3 inches forward of nominal from hover to 215 KTAS and in hover at 6 inches forward of nominal. No inherent problems were found.

Operational tests were conducted primarily in the nap-of-the-earth (NOE) and contour flight environments. This program was conducted at Fort Rucker, Alabama, in conjunction with the U.S. Army Aviation Development Test Activity.

The ABC's compact design and smaller rotor diameter was found to be an enhancing characteristic in confined areas. Qualitatively, downwash characteristics were comparable to the UH-60A. Contour flight at low altitudes was easily flown at speeds up to 215 knots. The full maneuver envelope was flown up to 10,000 feet density altitude (comparable to 6000 feet, 95°F condition). The maximum speed achieved at 10000 feet altitude was 240 KTAS.

Overall operational test results showed the Advancing Blade Concept to be a viable alternative for future tactical aircraft.

PREFACE

This report presents the results of two flight test programs conducted with the Advancing Blade Concept (ABC) demonstrator aircraft, serial number 21942. The first program evaluated maximum speed and altitude envelope expansion. The second was a Government test program conducted with the Army Aviation Development Test Activity in an operational environment. Program one was conducted by the Sikorsky Aicraft Division at the contractor's flight test facility, West Palm Beach, Florida. The second program was conducted at the contractor's facility and at Fort Rucker, Alabama. Both programs were conducted under contract DAAK51-80-C-0021 with the Applied Technology Laboratory (ATL), U. S. Army Research and Technology Laboratories, Fort Eustis, Virginia. A. Linden was the Sikorsky program manager.

Altitude flight testing in the auxiliary propulsion configuration commenced 19 August 1980 and was completed 20 January 1981. Sikorsky test pilots were B. Graham and D. Wright. Government evaluation pilots were D. Simon, ATL, LCDR T. MacDonald, Naval Air Test Center, and MAJ R. Carpenter, Army Aviation Engineering Flight Test Activity. Messrs. H. Young and D. Simon were the Army technical representatives for this program.

The Government test program began 27 April 1981 at the contractor's test facility and on 5 May 1981 at Fort Rucker, Alabama. It was completed 5 June 1981. Sikorsky test pilots were J. Dixson, B. Graham, N. Lappos, R. Murphy, and D. Wright. The initial government evaluation flight was made by D. Simon. Army Aviation Development Test Activity Pilots were Dr. J. Kishi, LTC R. Williams, MAJ J. Lash, and CW3 L. Scott. In addition, one evaluation flight in the auxiliary propulsion mode was made by COL E. F. Knight, ATL, and two were made by Capt. R. Rich, Navy Helicopter Antisubmarine Wing 1. D. Simon was the Army technical representative for the operational testing. Both programs were conducted under the supervision of A. Ruddell.

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INTRODUCTION

The Advancing Blade Concept (ABC) rotor system, employing a pair of counterrotating, coaxial, very rigid hingeless rotors, represents a significant departure from all predecessor helicopter rotor systems. It derives its name from the fact that the predominant lift load at high forward speeds is carried by the advancing blades on both sides of the aircraft. Since the retreating blades are not required to carry a significant fraction of the total lift load at forward speed, the speed and load factor limitations of the conventional helicopter due to retreating blade stall are eliminated. Unlike a conventional helicopter, rotor lift capability is retained with increasing speed, and speed capabiity is maintained at altitude.

In addition to performance benefits, the ABC's unique coaxial rigid rotors represent a significant departure from past practice in handling qualities, acoustics, loads, and dynamics. As with other coaxial counterrotating rotors, torque cancellation is provided, thereby eliminating the need for a tail rotor and its associated shafting and gearboxes.

Advancing Blade Concept development began in 1964. Extensive analytical and experimental studies culminated in the test of a 40-foot-diameter rotor in the Ames 40-x-80-foot wind tunnel in 1970 (Reference 1). The wind tunnel tests covered a speed range of 80 to 180 knots and advance ratios of 0.21 to 0.91. Test results confirmed the performance potential of the ABC rotor system. In addition, the full-scale wind-tunnel program developed materials technology and fabrication techniques to make construction of a demonstrator aircraft practical.

In December 1971 the U. S. Army awarded Sikorsky Aircraft a contract to design, fabricate, and fly the XH-59A to demonstrate the performance, handling qualities, and maneuver capabilities of the ABC rotor system. The initial technology demonstration, which included 66 flight hours in the helicopter mode and 40 hours in the auxiliary propulsion

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1. Paglino, Vincent M., and Beno, Edward A., FULL-SCALE WIND-TUNNEL INVESTIGATION OF THE ADVANCING BLADE CONCEPT ROTOR SYSTEM, Sikorsky Aircraft Division, United Aircraft Corporation; USAAMRDL Technical Report 71-25, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, August 1971, AD 734338.

mode, was completed 21 May 1980. The results are documented in Reference 2.

In June 1980 Contract DAAK51-80-C-0021 was awarded to continue envelope expansion with the XH-59A, including altitude, speed, maneuver, and center-of-gravity ranges.

The altitude testing was conducted in the auxiliary propulsion mode with the rotors indexed to cross each other at the 0-degree azimuth position. Tests were conducted at 10,000-, 16,000-, and 20,000-foot density altitudes. The aircraft established a service ceiling in excess of 25,000 feet density altitude.

The XH-59A demonstrator achieved a true airspeed of 240 KTAS in level flight at 10,000 feet and 263 KTAS in a shallow dive from 16,000 feet density altitude.

The forward center-of-gravity flight test was conducted at 3 inches and 6 inches forward of the nominal test center-of-gravity. This corresponded to 7 inches and 10 inches forward of the rotor shaft centerline. Ballast was carried on a structurally redesigned instrumentation nose boom. The ballasting technique was used to establish a sufficient moment and to maintain the aircraft within gross weight limits. The 6-inch-forward center-of-gravity flight testing was discontinued because of an insufficient structural margin in the ballasted instrumentation boom mounting structure to permit evasive or emergency maneuvering.

Tests were conducted in the auxiliary propulsion configuration at the Sikorsky West Palm Beach Test Center. The first altitude flight was conducted on 19 August 1980 and the program concluded on 20 January 1981. At the end of this phase of testing, the aircraft had completed a total of 111 flights for a total flight time of 128.6 flight hours.

Subsequently, contract DAAK51-80-C-0021 was modified to include flight tests in an operational environment in both the helicopter and auxiliary propulsion modes. This program was initiated 27 April 1981 with a limited 5-hour flight program to define the operating limits in the 0-degree rotor crossover modified helicopter configuration. The aircraft was then flown to Fort Rucker, Alabama for low-speed nap-of-the-earth (NOE) evaluations and contour flying at high and low speed. At the end of this phase, the aircraft had accumulated a total of 155.5 flight hours in 132 flights.

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2. Ruddell, A. J., et al., ADVANCING BLADE CONCEPT (ABC) TECHNOLOGY DEMONSTRATOR, Sikorsky Aircraft Division, United Technologies Corporation; USAAVRADCOM Technical Report 81-D-5, Applied Technology Laboratory, U. S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia, April 1981, AD A100181.

AIRCRAFT DESCRIPTION

The XH-59 Advancing Blade Concept demonstrator aircraft is designed as a research aircraft to investigate the rotor characteristics in both helicopter and auxiliary propulsion modes. Figure 1 shows the aircraft in the auxiliary propulsion mode. The modified helicopter configuration used in operational tests is shown in Figure 2. The primary difference in the conventional and modified helicopter modes is that the J-60 support beams remained installed in the modified mode. Table 1 summarizes the general aircraft attributes. Detailed design descriptions are presented in Reference 2.

TABLE 1. XH-59A AIRCRAFT ATTRIBUTES

Aircraft Length (rotor turning)	41 ft 8 in.
Fuselage Length	40 ft 10 in.
Main Landing Gear Tread	8 ft
Height	12 ft
Rotor Diameter	36 ft
Number of Rotors	2
Blades per Rotor	3
Rotor Separation	30 in.
Blade Taper Ratio	2:1 in.
Blade Twist (nonlinear)	-10 deg
Total Rotor solidity $\left(\frac{bc_{75}}{\pi R}\right)$	0.127
Precone Angle	3 deg
Prelag Angle	1.4 deg
Shaft Tilt	0 deg
Design Rotor Speed-helicopter	650 ft/sec
-aux propulsion	450 ft/sec
Drive System Design Power	1500 hp
Tail Surface - Horizontal	60 ft^2
- Vertical	30 ft^2
Elevator - Percent of Horizontal Tail	25
Rudder - Percent of Vertical Tail	30
Power Plants - Lift	(2) PT6-3
- Thrust	(2) J-60-P3A

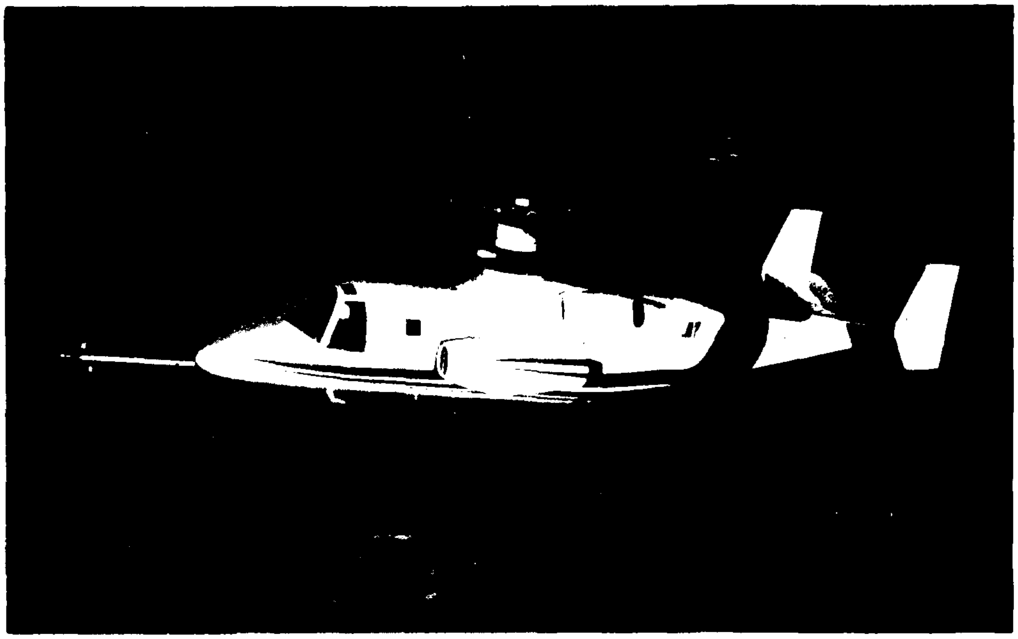

Figure 1. ABC Auxiliary Propulsion Configuration.

Figure 2. ABC Modified Helicopter Configuration.

ENVELOPE EXPANSION TEST RESULTS

FLIGHT ENVELOPE

During the altitude envelope expansion, specific Navy goals were established prior to the initiation of the test program. Table 2 lists these goals and the actual achievements.

TABLE 2. PROGRAM GOALS AND ACHIEVEMENTS

ITEM	GOAL	ACHIEVED
Maximum Altitude - ft	16,000	25,500
Maximum Speed - KTAS	250	263
Minimum Rotor Speed - %	80	78
High-Speed Maneuver	· Load Factor · Bank Angle · Roll Rates	240 kn 240 kn 220 kn

The 25,500-foot maximum altitude obtained was approximately the aircraft service ceiling (100 ft/min rate of climb). Above 25,000 feet a trim airspeed of 157 KTAS was sustained with a positive climb gradient.

The buildup to maximum airspeed was accomplished through three shallow dives. The first two were to 250 KTAS at 10,000, and 16,000 feet. The last was to 263 KTAS at 13,000 feet. At 263 KTAS, all rotating component stresses were less than working endurance values except the upper rotor shaft stress, which slightly exceeded its endurance value. This parameter directly reflected the amount of pitching moment being generated by the rotor. At this time, the aircraft was configured with an elevator locked at 2 degrees trailing edge down. As speed increased, tail lift increased, requiring an increase in nose-up rotor pitching moment to maintain trim. This led to reaching the upper shaft working endurance limits. A greater speed would probably have been achieved had program constraints allowed for the activation or repositioning of the elevator.

The full airspeed/altitude envelope explored is presented in Figure 3.

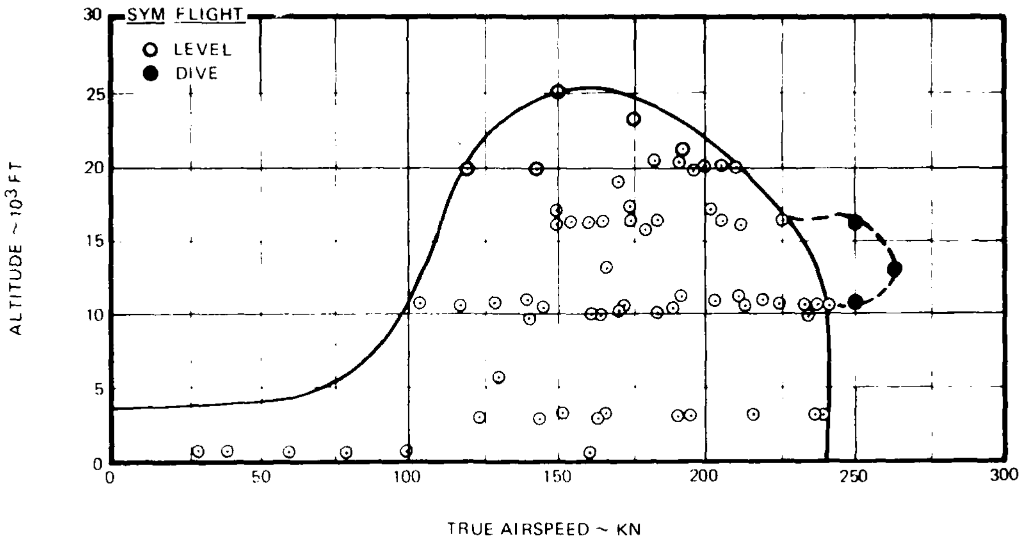

Figure 3. Altitude Envelope.

The minimum rotor speed demonstrated could only be achieved in the lower airspeed range (approximately 120 KTAS). With the present demonstrator rotor configuration and the rotor being flown in autorotation, significant rotor speed reduction could not be achieved at higher airspeeds. The rotor speed envelope and the constraints on it are discussed in detail in the structural test results section of this report.

A substantial maneuver envelope was opened at high speed throughout the altitude envelope. Figure 4 presents the aircraft V-n diagram. The bank angle envelope is shown in Figure 5.

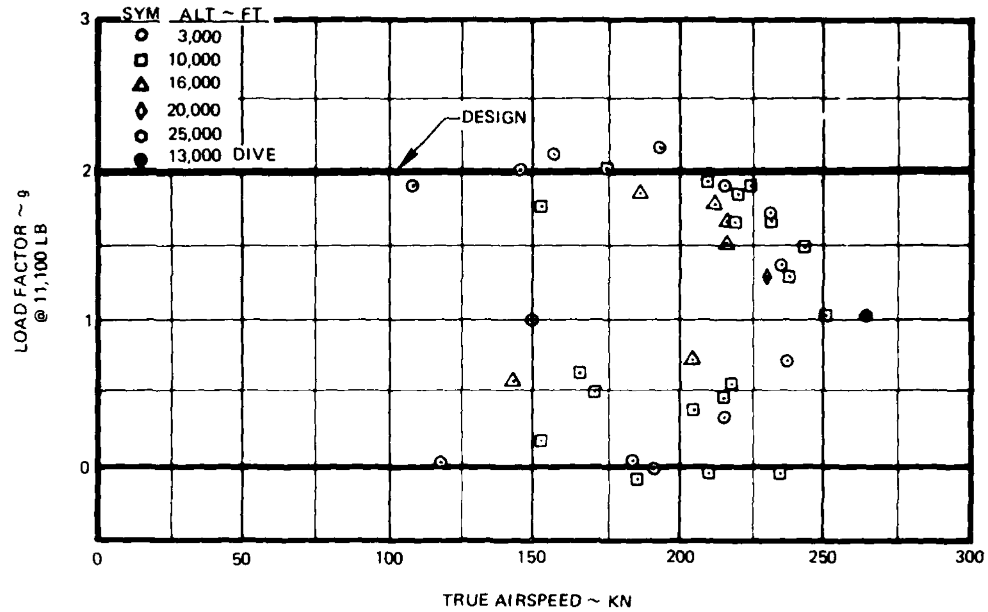

Figure 4. Load Factor Envelope.

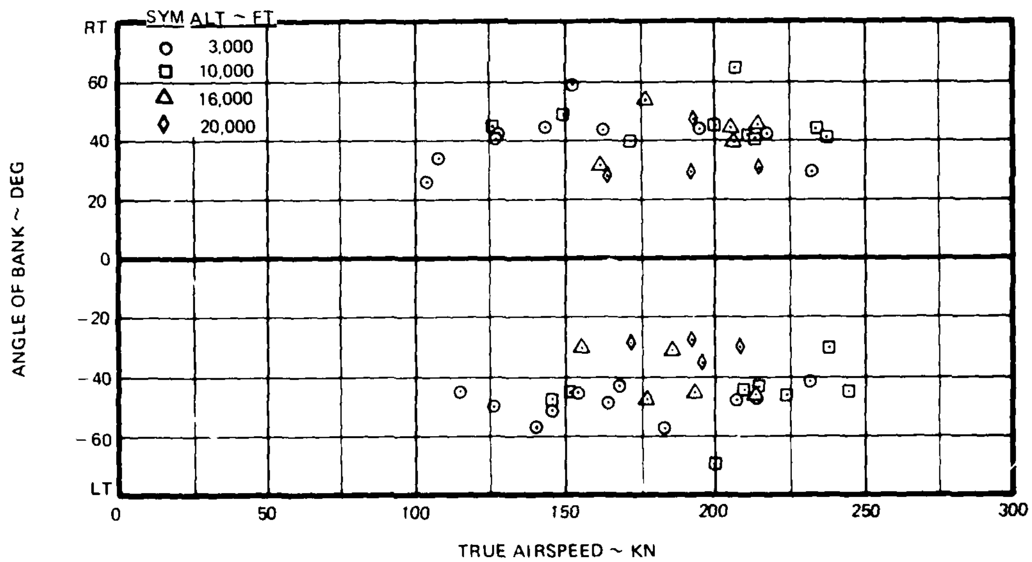

Figure 5. Bank Angle Envelope.

The high lift capability of the ABC rotor system is shown in nondimensional form in Figure 6. For a comparative base the UH-60A maneuver envelope has been included.

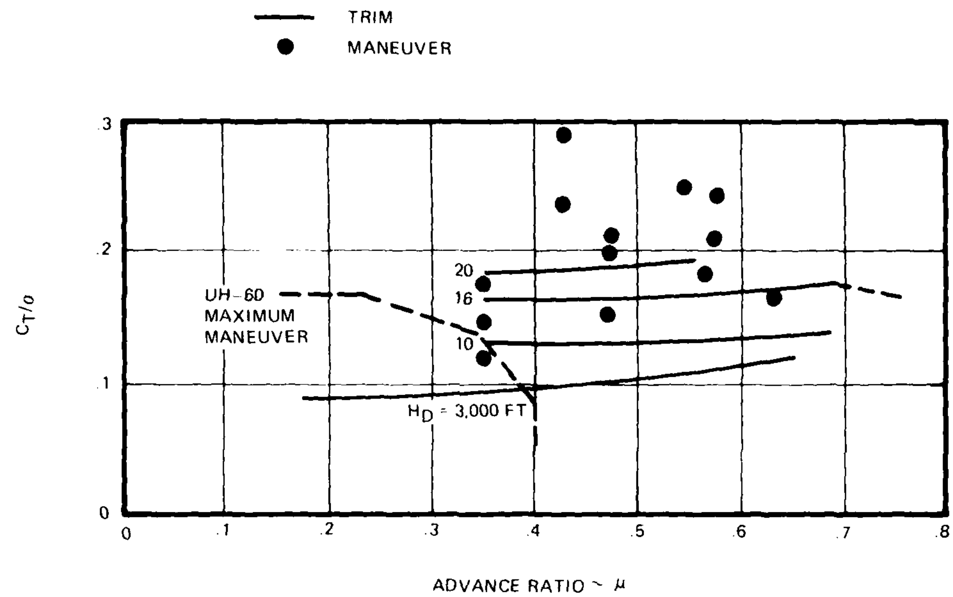

Figure 6. Maximum Nondimensional Blade Loading Demonstrated.

The demonstrated sideslip envelope is presented in Figure 7. The reduction in sideslip angle above 200 knots is due to rudder pedal force requirements to maintain a given sideslip angle.

The rate of climb and descent envelope developed are shown in Figure 8. Higher rates of climb were not achieved at lower airspeeds due to a pitch attitude limit of +20 degrees. This restriction was applied to avoid cavitation of the gearbox oil pump.

Level acceleration from a hover was also evaluated. Figure 9 shows a typical time history of one of these accelerations.

PERFORMANCE

The effect of altitude on 1 g flight performance is shown in Figure 10. Increased altitude is shown in terms of increased referred gross weight (W/δ) as shown in Table 3.

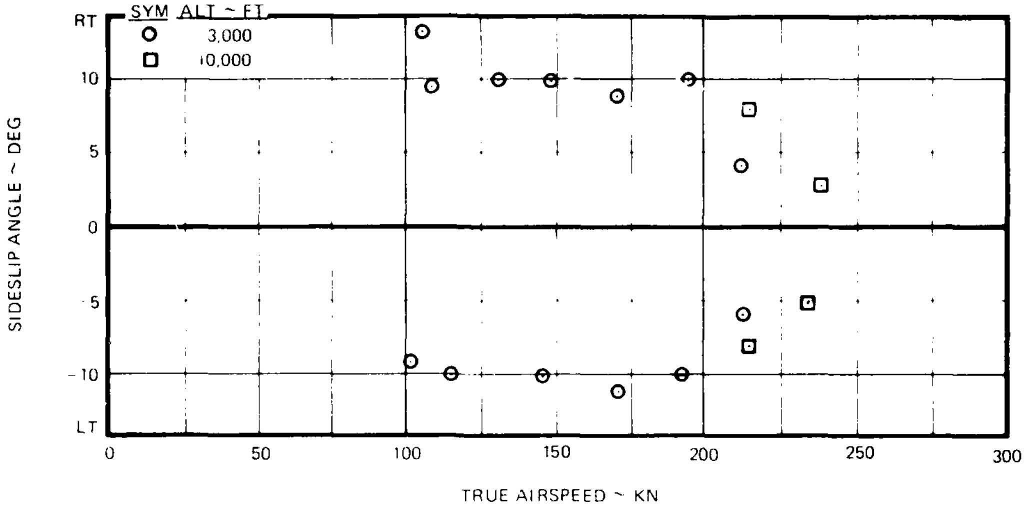

Figure 7. Sideslip Envelope.

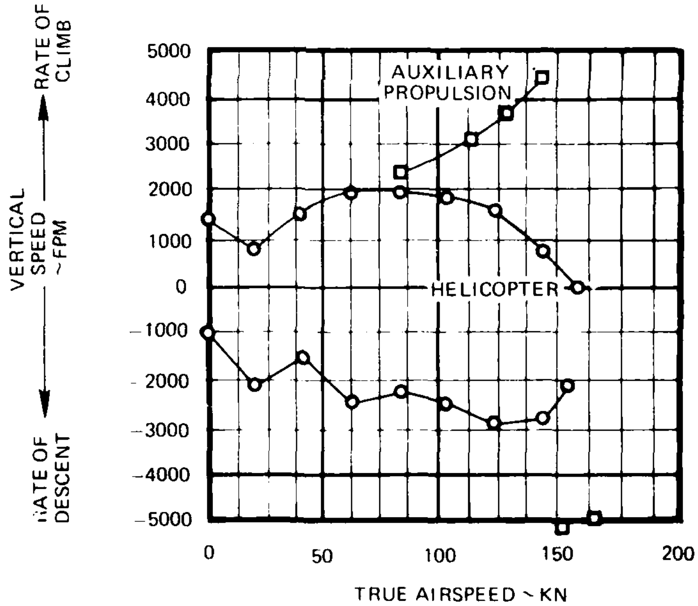

Figure 8. Rate of Climb and Descent Envelope.

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B

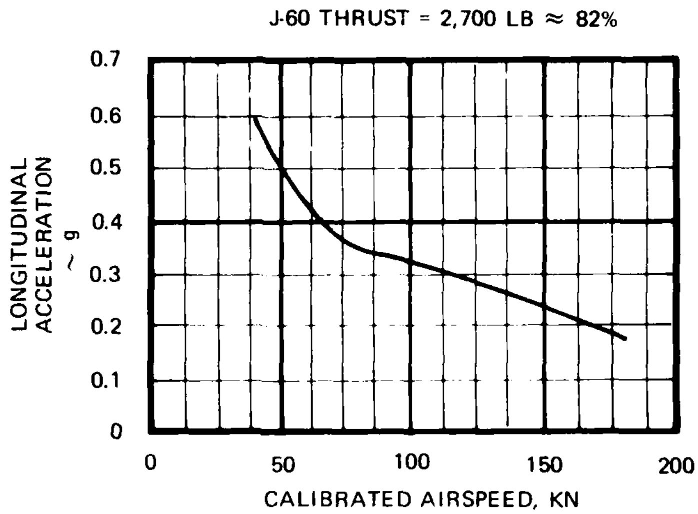

Figure 9. Level Acceleration Demonstrated.

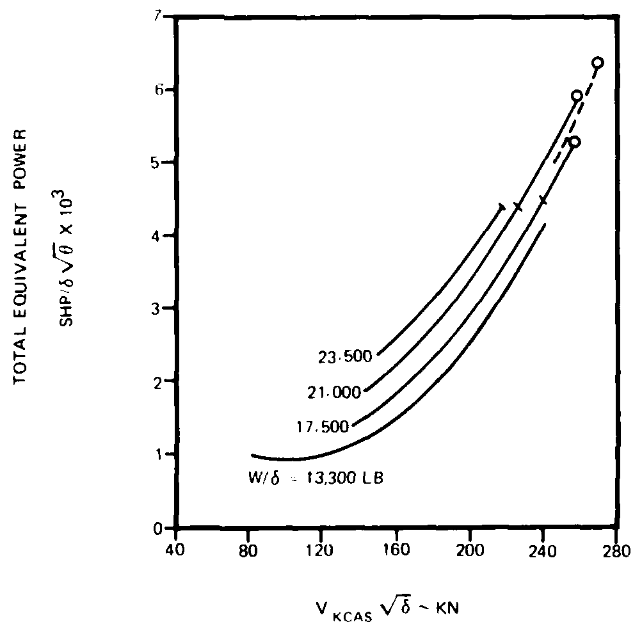

Figure 10. 1 g Flight Performance.

TABLE 3. DENSITY ALTITUDE/REFERRED GROSS WEIGHT

Density Altitude - ft	Referred Gross Weight - lb
3,000	13,300
10,000	17,500
16,000	21,000
20,000	23,500

In Figure 10, each of the dive points at 10,000, 13,000, and 16,000 feet is identified by a data symbol. The V_H achieved at each altitude is identified by a tick mark.

HANDLING QUALITIES

Flight testing of the auxiliary propulsion configuration was conducted to expand the altitude envelope and the forward center-of-gravity range. Previous testing showed that stability and handling qualities of the XH-59A in the auxiliary propulsion configuration were excellent with no control margin limitations or unstable modes apparent. Test planning was directed to evaluate aircraft structural and vibratory characteristics during envelope expansion. Handling qualities data presented and discussed are not the results of dedicated handling qualities flights. Testing was conducted with the upper and lower rotors oriented in 0-degree crossover and the elevator set at 2 degrees trailing edge down.

Takeoff and Climb

Takeoff techniques had been developed in tests conducted at Rentschler Field, East Hartford, Connecticut, and are reported in Reference 2. The same takeoff procedures were used throughout this test phase. The prime emphasis was to develop a fuel-efficient climb profile to maximize test time at altitude. Investigation of climb data showed that the fuel-efficient climb profile of the XH-59A was bounded by two constraints. The first, in the initial portion of the climb, was lower rotor blade stress. As collective was increased to increase rate of climb, the lower rotor blade structural endurance limit was reached. As altitude increased, the lower blade structural endurance level was no longer constraining. The second constraint was forward longitudinal control margin. As collective control was increased, the longitudinal control migrated forward. It was found that above 10,000 feet the collective control position was limited by a forward longitudinal 10-percent control margin of safety.

The following climb profile was used throughout the altitude test program:

- Maintain rotor speed at 102 percent throughout takeoff and climb.
- Takeoff - collective 50 percent, J-60 thrust 30 percent increasing to maximum.
- Initial Climb - collective 50 percent, attitude 20 degrees nose up.
- Climb 3,000 to 7,000 feet - collective 55 percent, attitude 20 degrees nose up.
- Climb 7,000 to 18,000 feet - increase collective until 10 percent longitudinal control is remaining.
- Climb 18,000 feet and above - maintain collective and fly calibrated airspeed between 95 and 100 knots.

This climb technique was used to achieve the XH-59A service ceiling in excess of 25,000 feet density altitude. Figure 11 presents the parameters for the climb to service ceiling.

Descents

The prime emphasis placed on aircraft descent was to verify that the emergency descent procedure developed from 3,000 feet density altitude would remain valid from any altitude.

The descent was initiated by bringing the J-60 thrust engines back to idle, or securing the engines to conserve fuel and setting a collective control position. The aircraft attitude was adjusted to hold the desired calibrated airspeed. Figure 12 presents descent data for a target 40-percent collective setting. As calibrated airspeed was increased, the rate of descent increased and approached 5,000 feet per minute at 155 KCAS. There were no structural limitations in the envelope shown. The aircraft was easily trimmed and handling qualities were considered excellent in descents and descending turns.

Controllability in Level Flight

Trim level-flight controllability data were generated from 120 KCAS to 200 KCAS, or V_H for 3,000-, 10,000-, 16,000-, 20,000-, and 25,000-foot density altitudes. All points were flown in accordance with the collective/B_1' schedule developed to maintain the upper rotor hub stress at endurance (see Figure 13). Figure 14 is a composite plot of level flight controllability for all test altitudes. Examination of the data

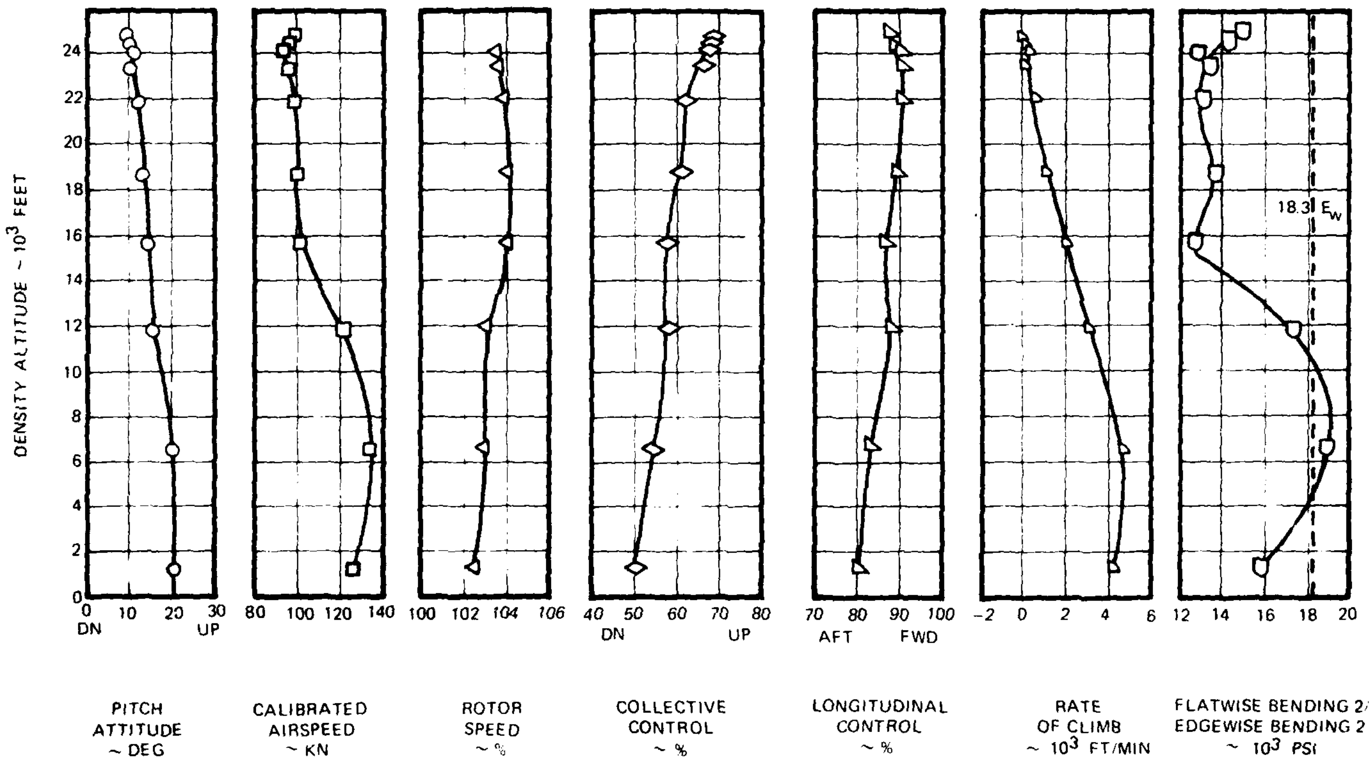

Figure 11. Climb Profile - Climb to Service Ceiling.

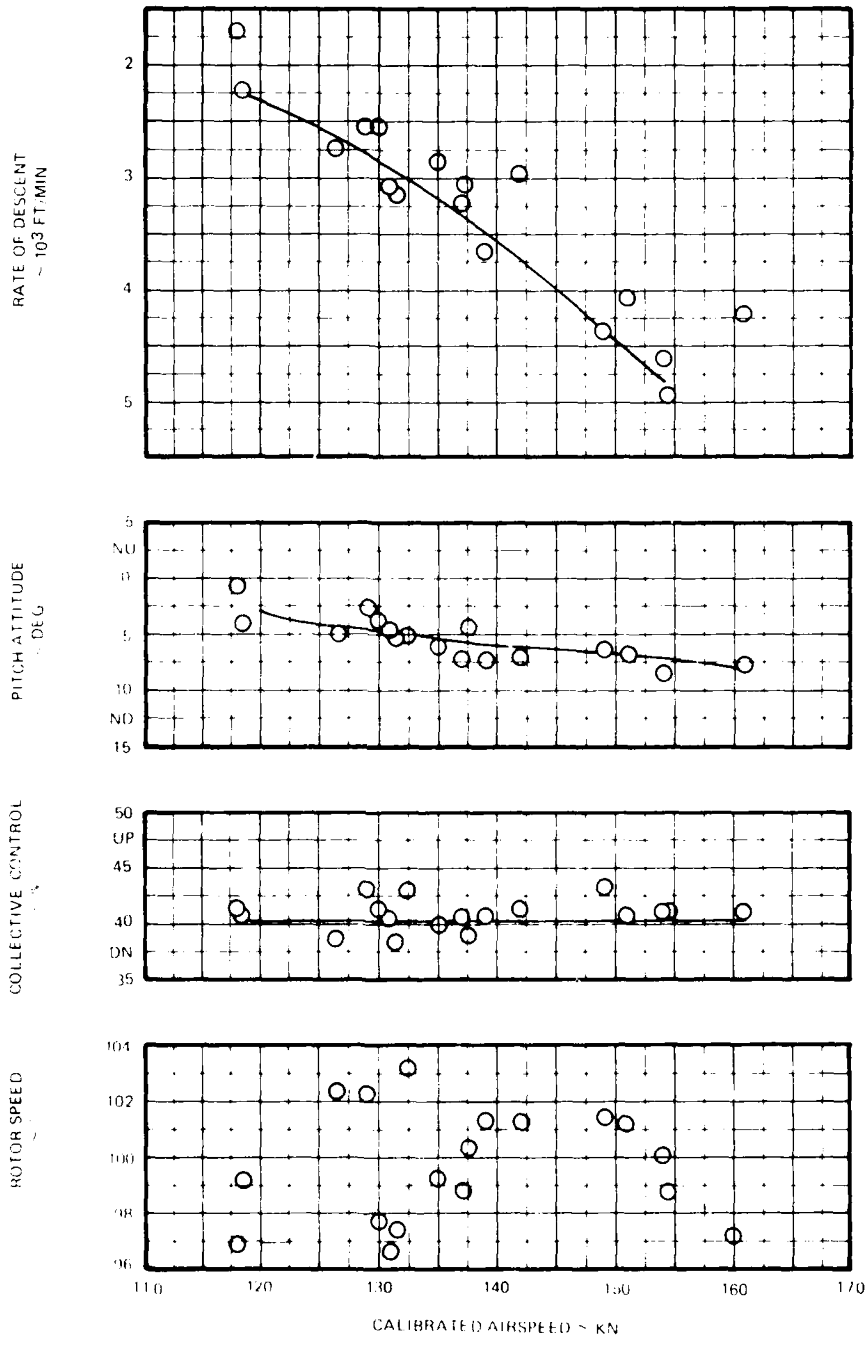

Figure 12. Descent Profiles.

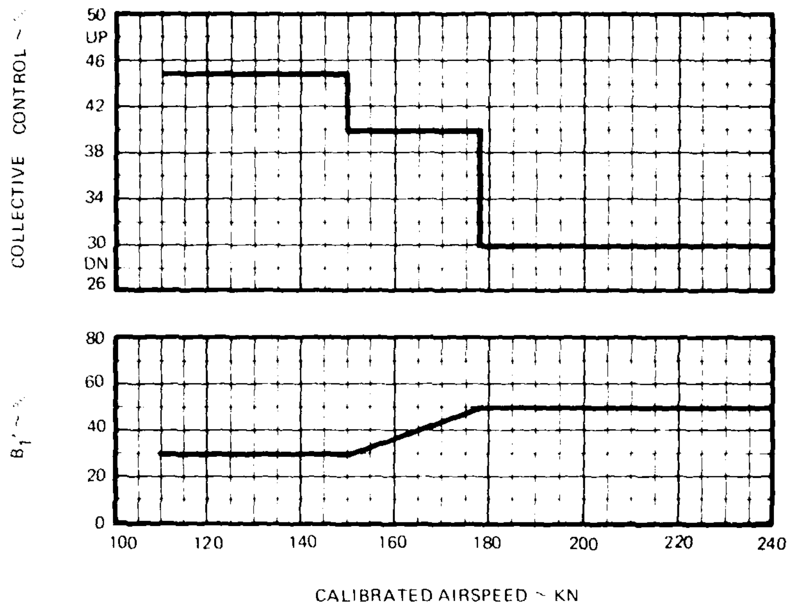

Figure 13. Cockpit Control Schedule.

shows that density altitude has no significant effect on pitch attitude or flight control positions when plotted against calibrated airspeed. For a constant collective control position, the longitudinal control gradient is positive with increasing airspeed for all density altitudes. The longitudinal control gradients for the three scheduled collective control positions are identified. The relationship of longitudinal position to collective position is presented in Figure 15. Lateral and directional control positions were consistent at all density altitudes. Handling qualities of the basic aircraft were excellent throughout, with the entire envelope being flown with SAS on and off.

Figure 16 presents angle of attack versus calibrated airspeed with all data normalized to a collective position of 40 percent. Here it can be seen that there was no discernable effect of altitude on aircraft angle of attack at the same calibrated airspeed.

Differential Control Effects

The effects of changes in differential longitudinal (A_1') and differential collective trim ($\Delta\theta_t$) were evaluated at 10,000 feet. Figure 17 presents a plot of pitch attitude and the four flight control positions for changes in A_1'. Changes in differential longitudinal control had no effect on aircraft attitude or flight control positions.

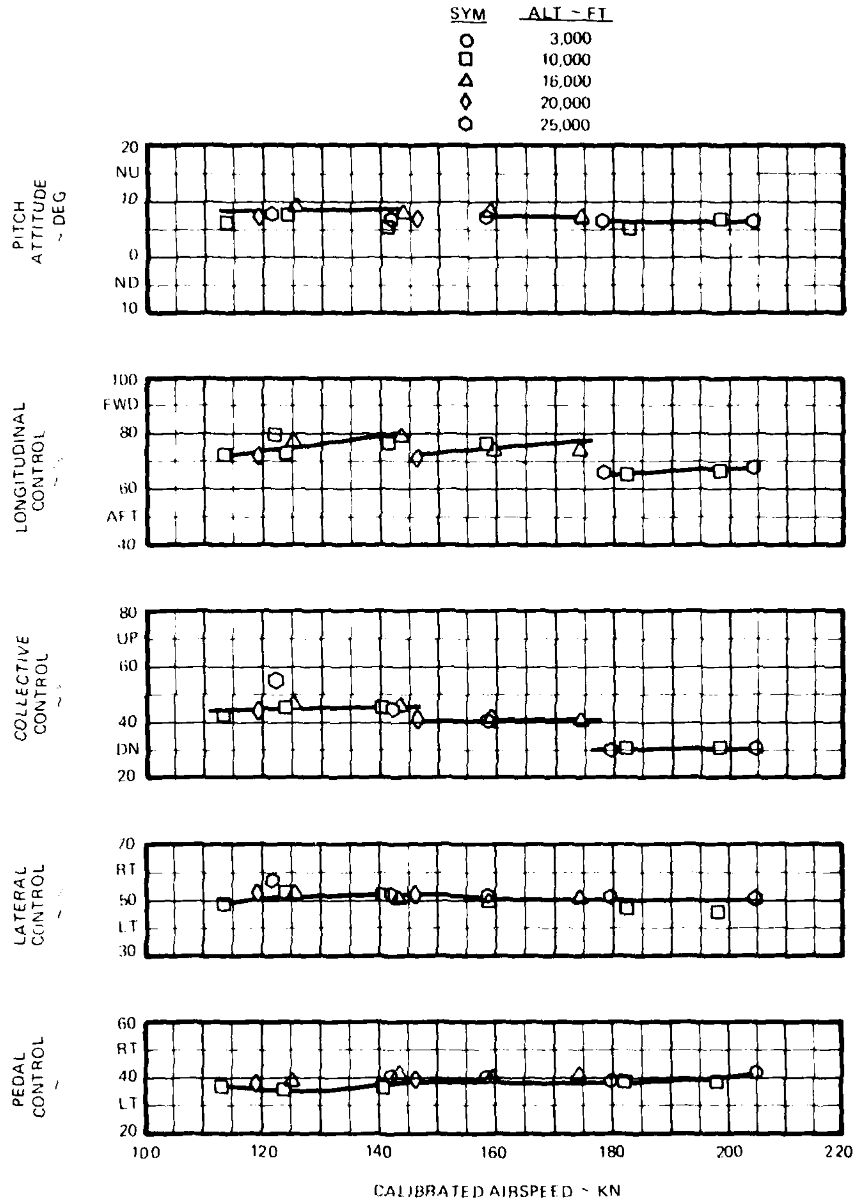

Figure 14. Level Flight Controllability.

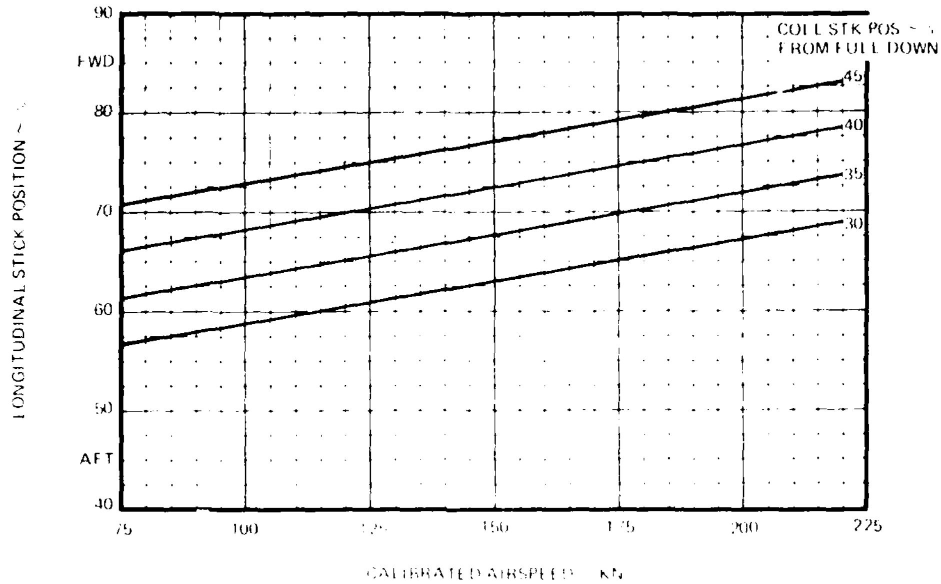

Figure 15. Collective/Longitudinal Control Relationship.

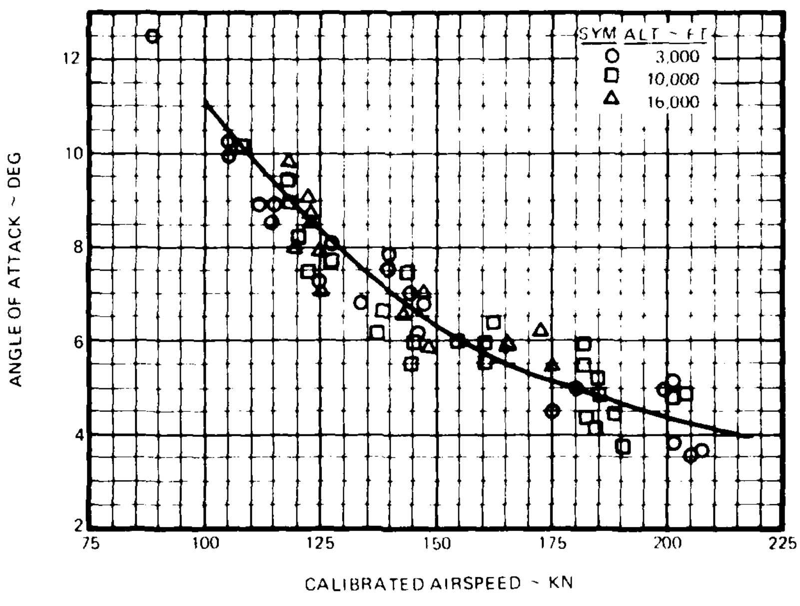

Figure 16. Normalized Angle of Attack vs Airspeed.

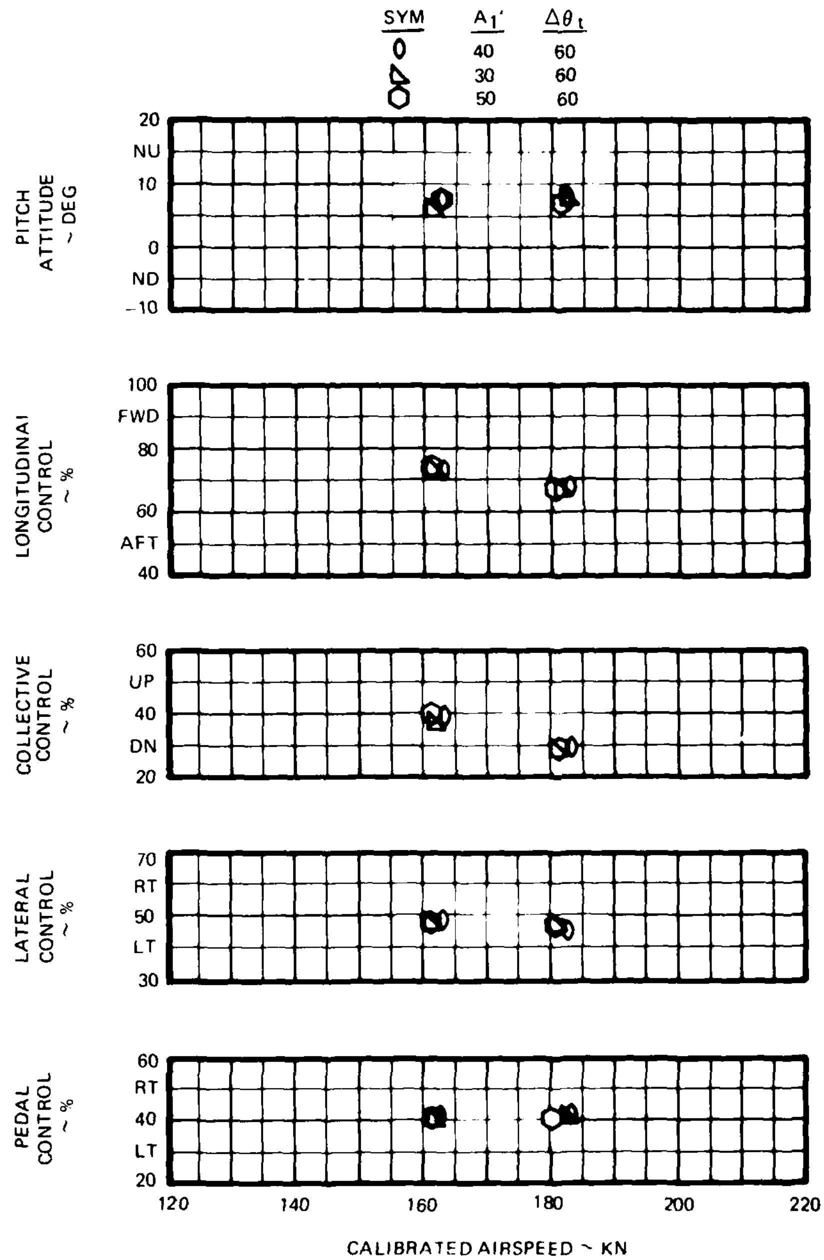

Figure 17. Level Flight Controllability - Effect of Differential Longitudinal Control.

Figure 18 presents a plot of pitch attitude and the four flight control positions for changes in $\Delta\theta_t$. Changes in differential collective control influenced the lateral control position. As the differential collective control was increased, the lateral control migrated to the right. Increasing differential collective control increases the collective pitch on the upper rotor and decreases collective pitch on the lower rotor. Increasing collective pitch on the upper rotor increased the lift vector on the advancing right side; decreasing the collective pitch on the lower rotor decreased the lift vector on the advancing left side. To compensate for the change in the lift vectors, right lateral control was introducted to balance the rolling moments. Data for 160 and 180 KCAS showed that the relationship between lateral control and differential collective control is 0.2 percent right lateral control displacement per percent increase in differential collective control.

Level-Flight Controllability Center-of-Gravity Effects

Forward center-of-gravity flight testing was conducted 3 inches and 6 inches forward of the nominal center of gravity (296 inches), at 3,000 feet density altitude. Trim level flight was conducted to 205 KCAS for the 3-inch forward center-of-gravity condition and hover flight was evaluated for the 6-inch forward center-of-gravity condition. The 6-inch forward center-of-gravity test was terminated at low-speed flight because of inadequate structural margins in the nose boom for forward flight evasive maneuvering. The nose boom was used to attach the ballast to establish forward center of gravity and maintain the aircraft within gross weight limits.

Figure 19 presents the hover and level-flight controllability plot for the center-of-gravity testing. For a rigid rotor, small changes in pitch attitude are expected. In hover the aircraft demonstrated less than 3 degrees change in pitch attitude for a 6-inch change in center of gravity. The longitudinal control position should move aft as the center of gravity moves forward. This trend in the hover testing appeared to be reversed. The longitudinal control position for the 3-inch forward center-of-gravity test fell within 0.4 percent of the nominal center-of-gravity test. The 6-inch forward center-of-gravity condition fell 1.1 percent forward of the nominal center-of-gravity longitudinal control position. The 6-inch center-of-gravity hover data was recorded in winds up to 12 knots. Wind conditions for all other configurations were less than 3 knots.

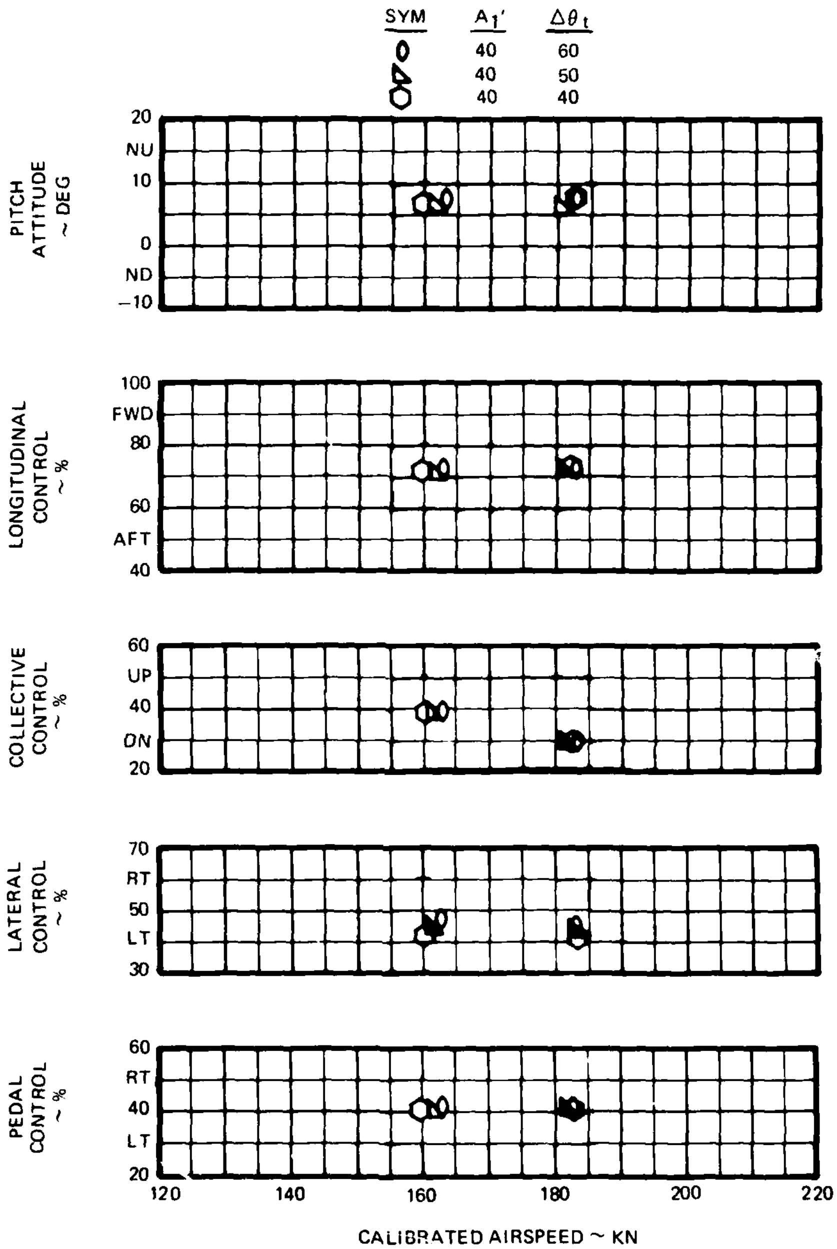

Figure 18. Level Flight Controllability - Effect of Differential Collective Trim.

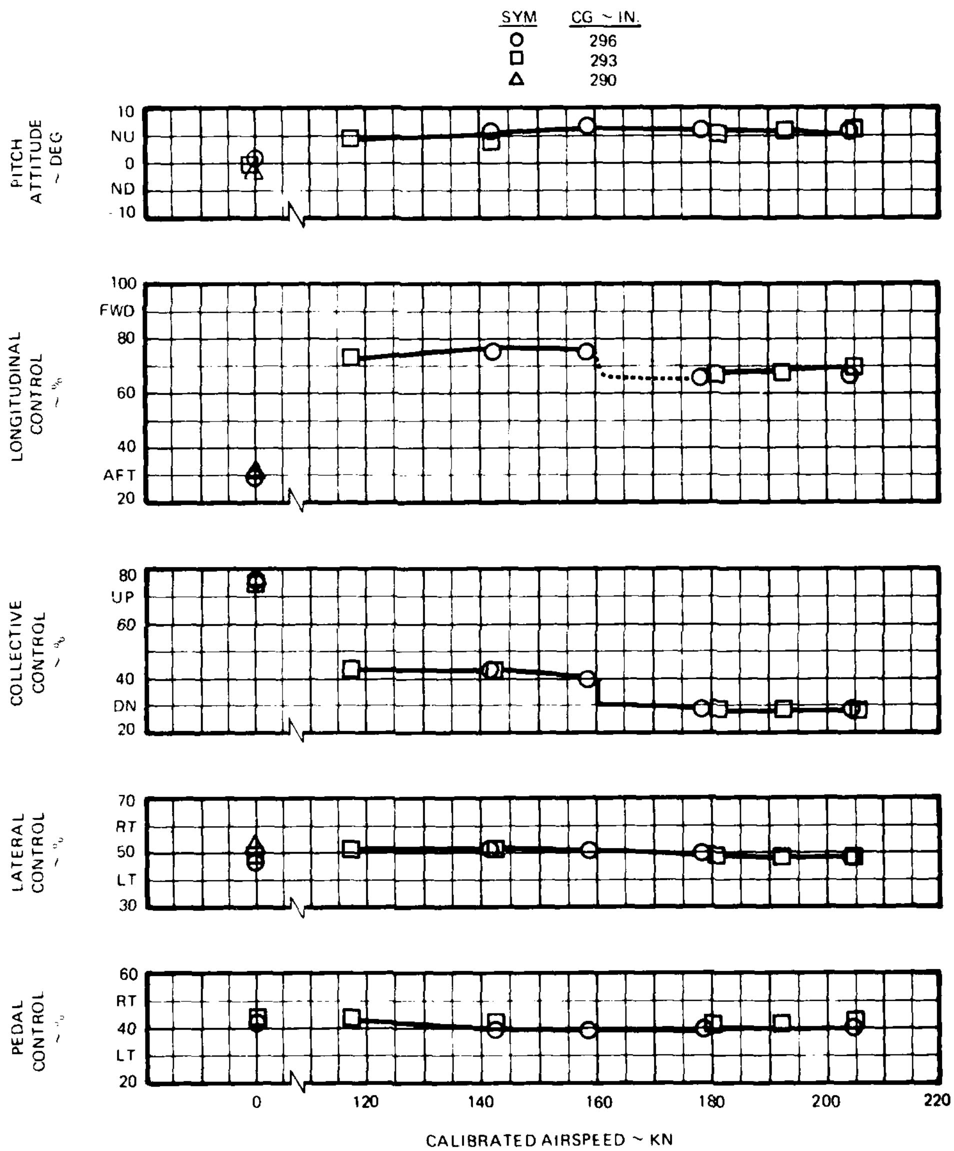

Figure 19. Level Flight Controllability - Effect of Center-of-Gravity Location.

In forward flight the nominal center of gravity pitch attitude was slightly more nose up than the 3-inch forward center of gravity data. As speed increased the aircraft attitudes converged, and at speeds above 180 KCAS there is no significant change. In level flight, control positions were repetitive. As would be expected with a rigid rotor system, the changes in flight control positions were imperceptible with respect to changes in longitudinal center of gravity.

Turns

Bank turns to right and left were conducted at various airspeeds during altitude and forward center of gravity envelope expansion. Piloting technique was to set collective control position and J-60 throttle position to maintain trim level flight at an airspeed. The aircraft was rolled to a constant bank-angle turn, exchanging altitude to maintain airspeed. Plots of constant turns at various altitudes and constant calibrated airspeed are presented in Figures 20 through 24. Trending curves have been faired through the data. Just as with the level flight controllability data, the flight control position data for constant turns fell within a reasonable scatter band, and were independent of altitude.

As turns were executed either left or right, the longitudinal control position moved aft in a stable maneuvering direction. Lateral control position remained neutrally stable or moved in a stable maneuvering direction. At 120 KCAS, right pedal control was required to trim in a left bank. Pedal control for a right bank was in a stable direction. At 160 KCAS the pedal control positions were neutrally stable. At higher airspeeds, pedal control position was in a stable direction for left turns and an unstable direction for right turns. Left pedal control was required for left or right turns at 140, 180, and 200 KCAS. This same phenomenon was observed during a symmetrical pullout. The aircraft would yaw right, requiring left pedal. This suggests a possible change in the rotor wake impingement on the vertical stabilizers.

Lateral Directional Static Stability

Lateral directional static stability was evaluated at 10,000 feet density altitude and 140, 180, and 200 KCAS. Piloting technique was to set collective position in accordance with the aircraft control schedule and adjust the J-60 throttles to maintain the trim level airspeed. The aircraft was sideslipped while maintaining a constant heading and altitude was exchanged to maintain airspeed. The data

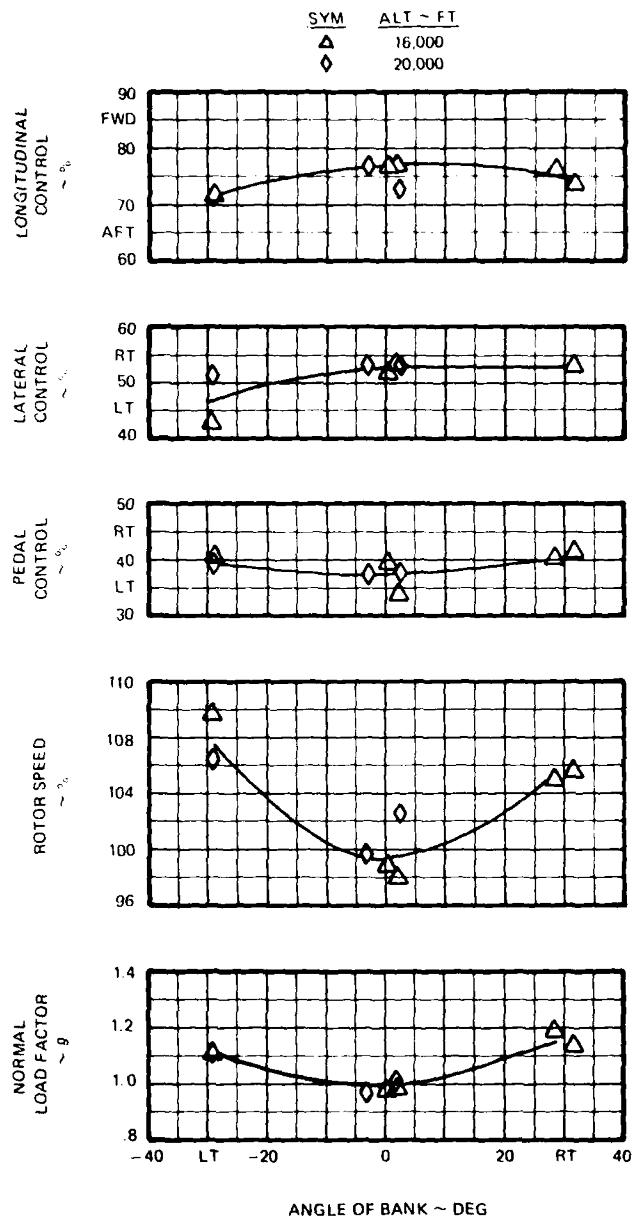

Figure 20. Turn Characteristics - 120 KCAS.

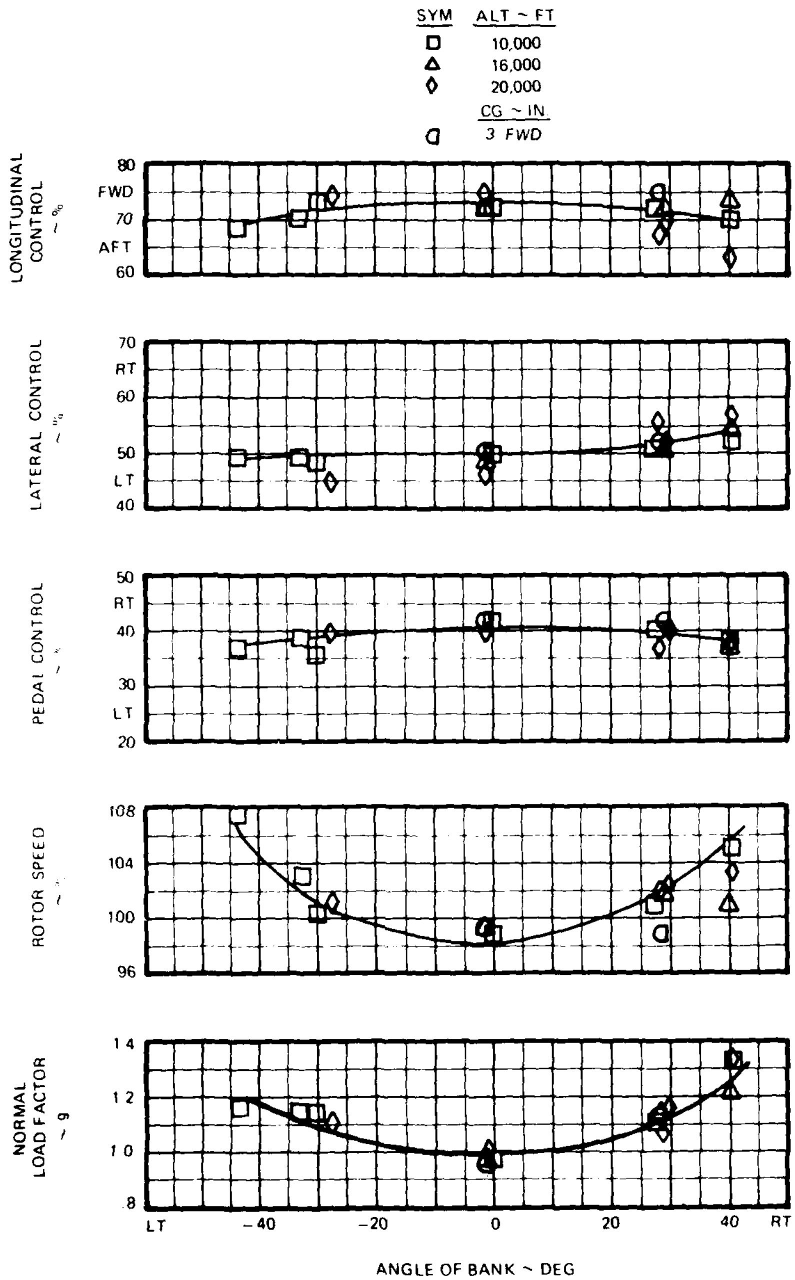

Figure 21. Turn Characteristics - 140 KCAS.

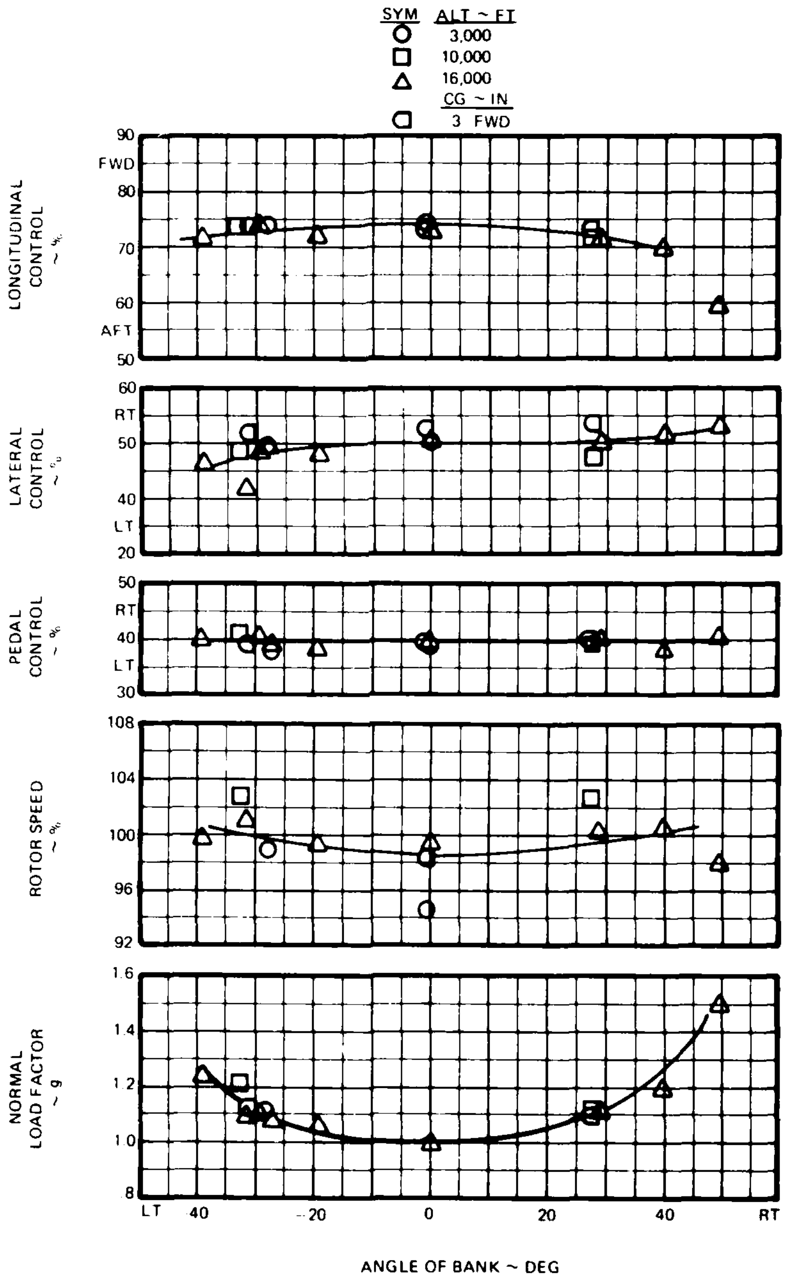

Figure 22. Turn Characteristics - 160 KCAS.

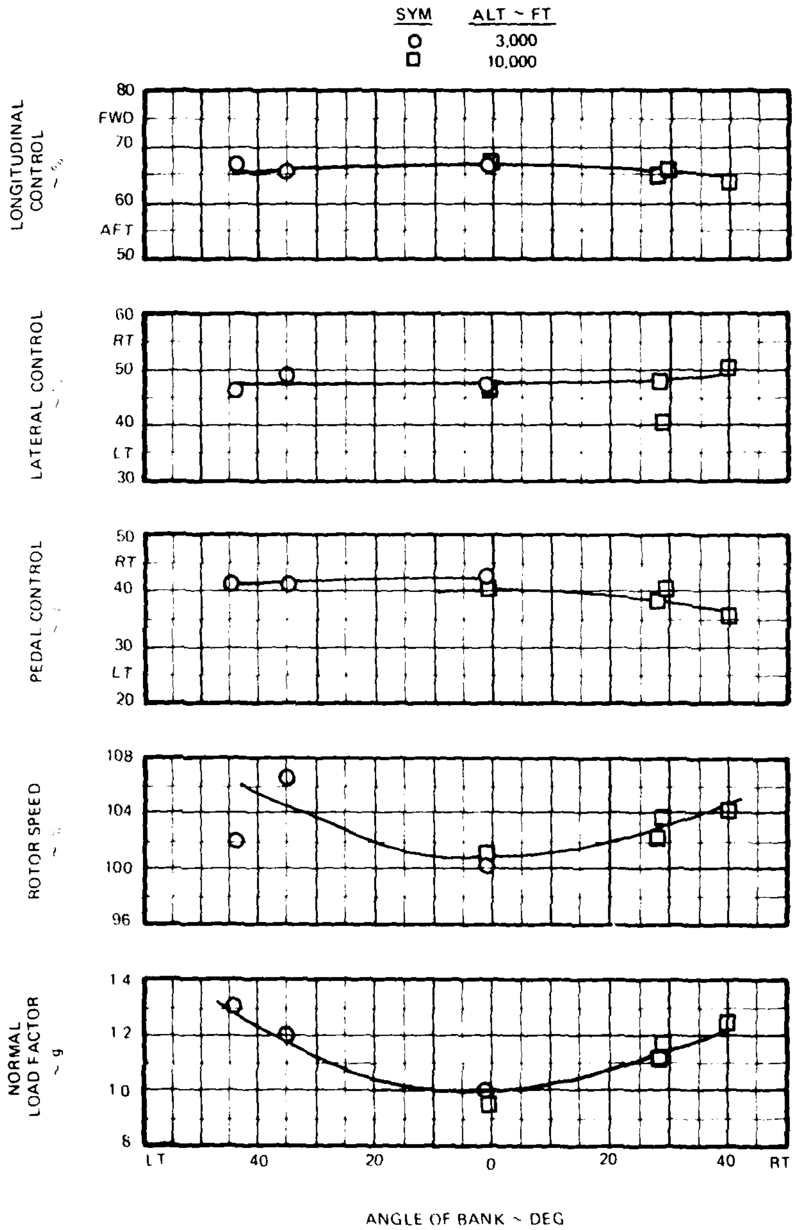

Figure 23. Turn Characteristics - 180 KCAS.

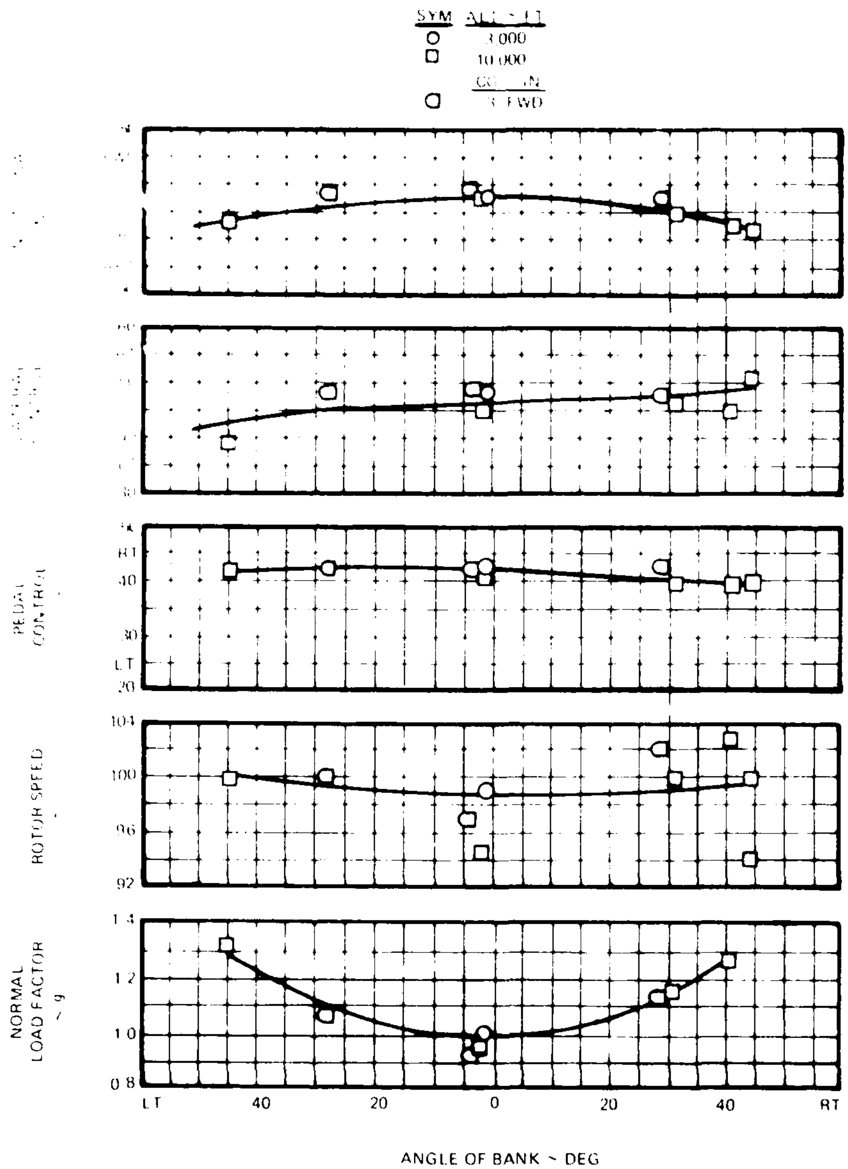

Figure 24. Turn Characteristics - 200 KCAS.

presented in Figure 25 show that the XH-59A exhibits positive lateral directional static stability characteristics. Angle of bank and pedal control displacement with sideslip angles were linear and independent of airspeed. Examination of the data showed that the static directional stability characteristics were independent over the airspeed range tested and varied linearly with respect to sideslip. The lateral control position data showed that the aircraft trimmed with a different lateral control position for zero sideslip. This is indicative of a change in dihedral effect with respect to airspeed.

Control Sensitivity

Control sensitivity was evaluated with respect to changes in airspeed and altitude. Flight control step inputs were introduced in each axis and the angular acceleration was recorded. The data are presented in Figure 26. Comparison of the data for the 3,000- and 10,000-foot density altitudes showed no significant effects with altitude and were within normal scatter.

Asymmetric Loss of Thrust

Loss of the J-60 thrust engines was simulated at altitudes up to 20,000 feet. The results were similar to those obtained at nominal sea level. Attitude excursions were mild and minimal control inputs were required to maintain the desired flight path.

Symmetric or asymmetric loss of thrust in any flight regime was considered benign and delay time prior to the pilot-required action was long. The primary effect was loss of speed.

STRUCTURAL RESULTS

Structural tests were conducted at density altitudes of 10,000, 16,000, and 20,000 feet and at the service ceiling of 25,500 feet. A small portion of testing was done at 3,000 feet to investigate center-of-gravity changes and to establish a data base with the elevator strapped at 2 degrees trailing edge down. The 2-degree elevator angle and the 0-degree rotor crossovers were maintained throughout the test program.

Initially, it was believed that rotor performance and loads were a function of true airspeed. The 10,000- and 16,000-foot initial flight envelope was flown using the cockpit control schedule of Reference 2 converted to true airspeed.

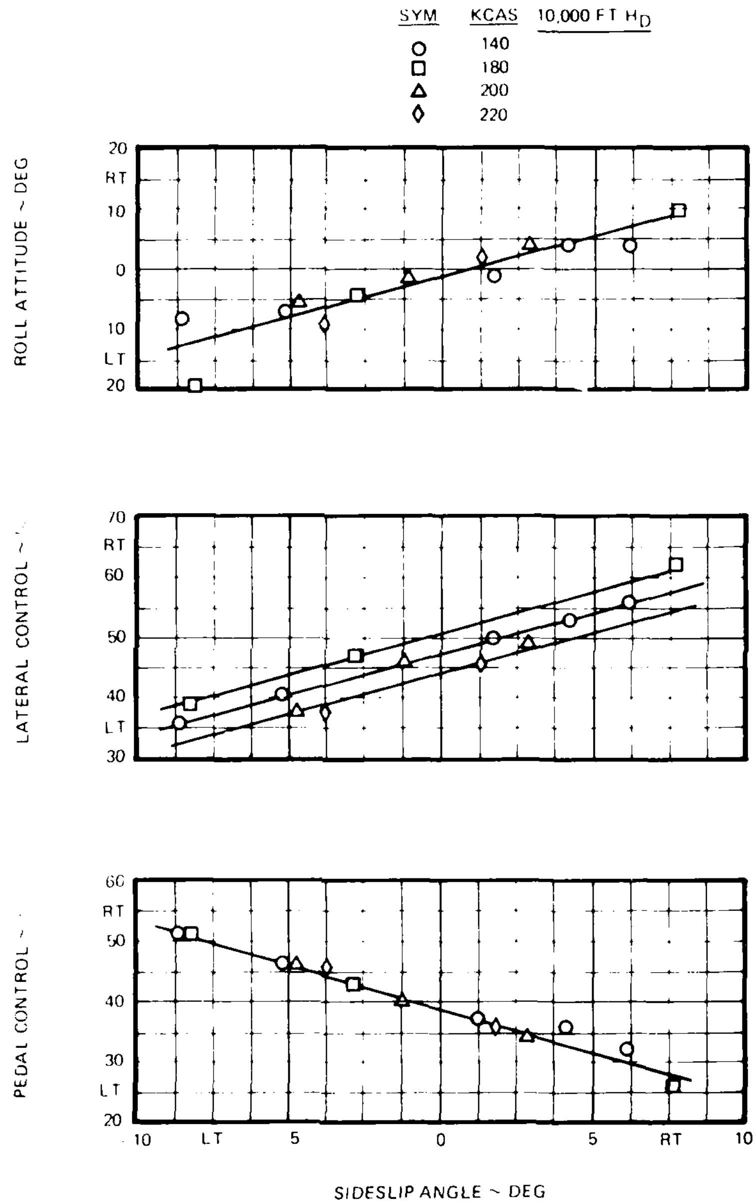

Figure 25. Sideslip Characteristics.

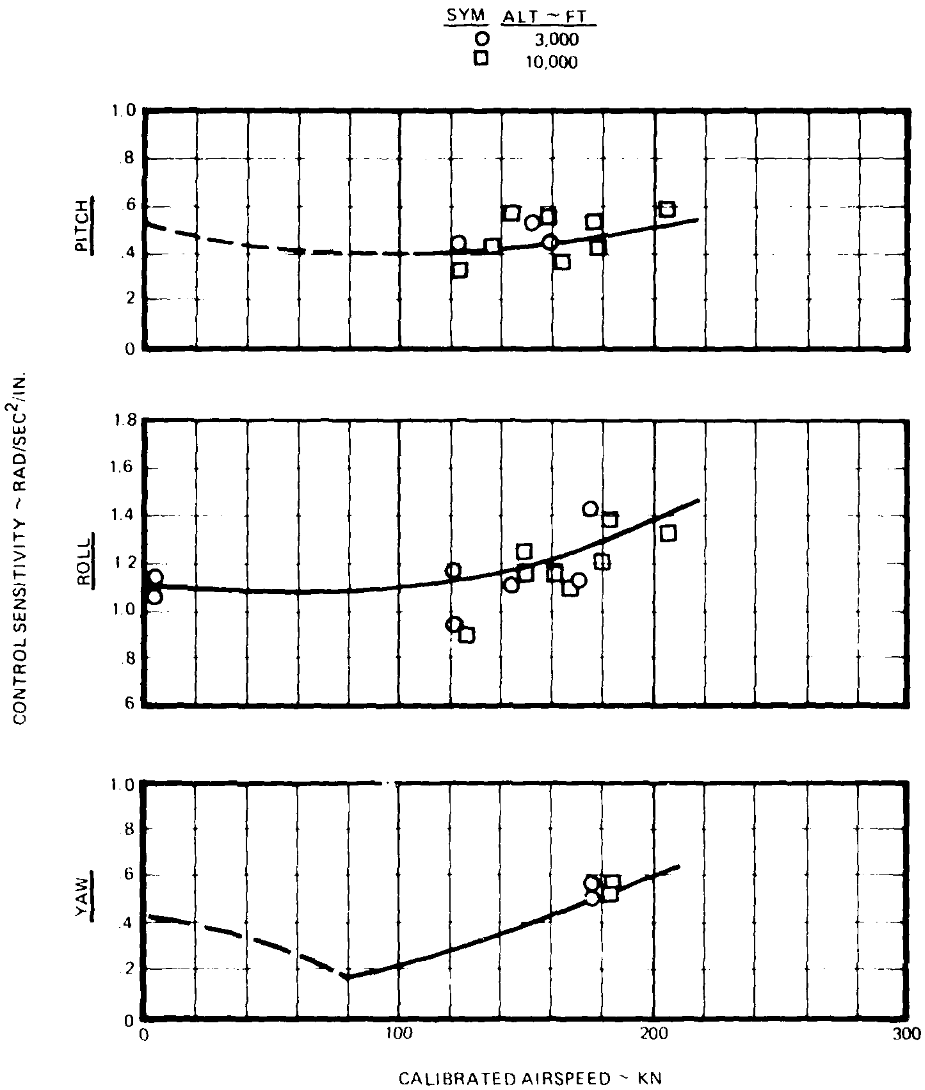

Figure 26. Control Sensitivity.

It became apparent that the aircraft would have to be flown in a more aerodynamically efficient manner if the high-speed potential of the XH-59A was to be evaluated at altitude.

Also, as experience was gained in expanding the envelope, it became apparent that the PT6 torque measuring system was inaccurate at very low values. In fact, a zero torque indication could be obtained while power was being supplied to the rotor system. There was a significant effect on rotor loads and stress as a function of the rotor torque being applied. To ensure consistency in the data, the rotor was purposely flown in autorotation by the technique of splitting the power turbine and rotor speed needles on the pilot's tachometer. This was accomplished by moving the PT6 power levers to the ground idle position once level flight was achieved.

Only data for the rotor parameters are presented in this report. The airframe stress characteristics were primarily affected by rotor crossover as presented in Reference 2. The rotor crossover remained at 0 degrees for the data presented herein, and the only apparent change in airframe stress with altitude was a shift in tailcone maximum stress from the tailcone/fuselage attachment to the aft tailcone area.

Effects Of Differential Lateral Control (B_1')

Increasing differential lateral control decreased blade pitch on the advancing side of each rotor and increased the blade pitch on the retreating side of each rotor. B_1' effects on the rotor for 3,000 feet density altitude have been presented in Reference 2. Since no attempt was previously made to keep the rotor in autorotation, and since altitude effects were uncertain, B_1' variations were flown at 10,000 and 16,000 feet. An attempt was made to investigate B_1' variations at 20,000 feet; however, due to the limited fuel capacity of the XH-59A, it was impractical to acquire sufficient data. The effects of B_1' were then spot checked at 3,000 feet density altitude at 180 and 200 KCAS.

The data from the various altitude tests produced results similar to the data of Reference 2. However, to investigate the effects of B_1' with altitude, the derivatives of the various rotor parameters with respect to B_1' were calculated and are presented in Figure 27.

The results of the test indicate the following:

1. The effect of B_1' on the rotor load/stress did not change with altitude.

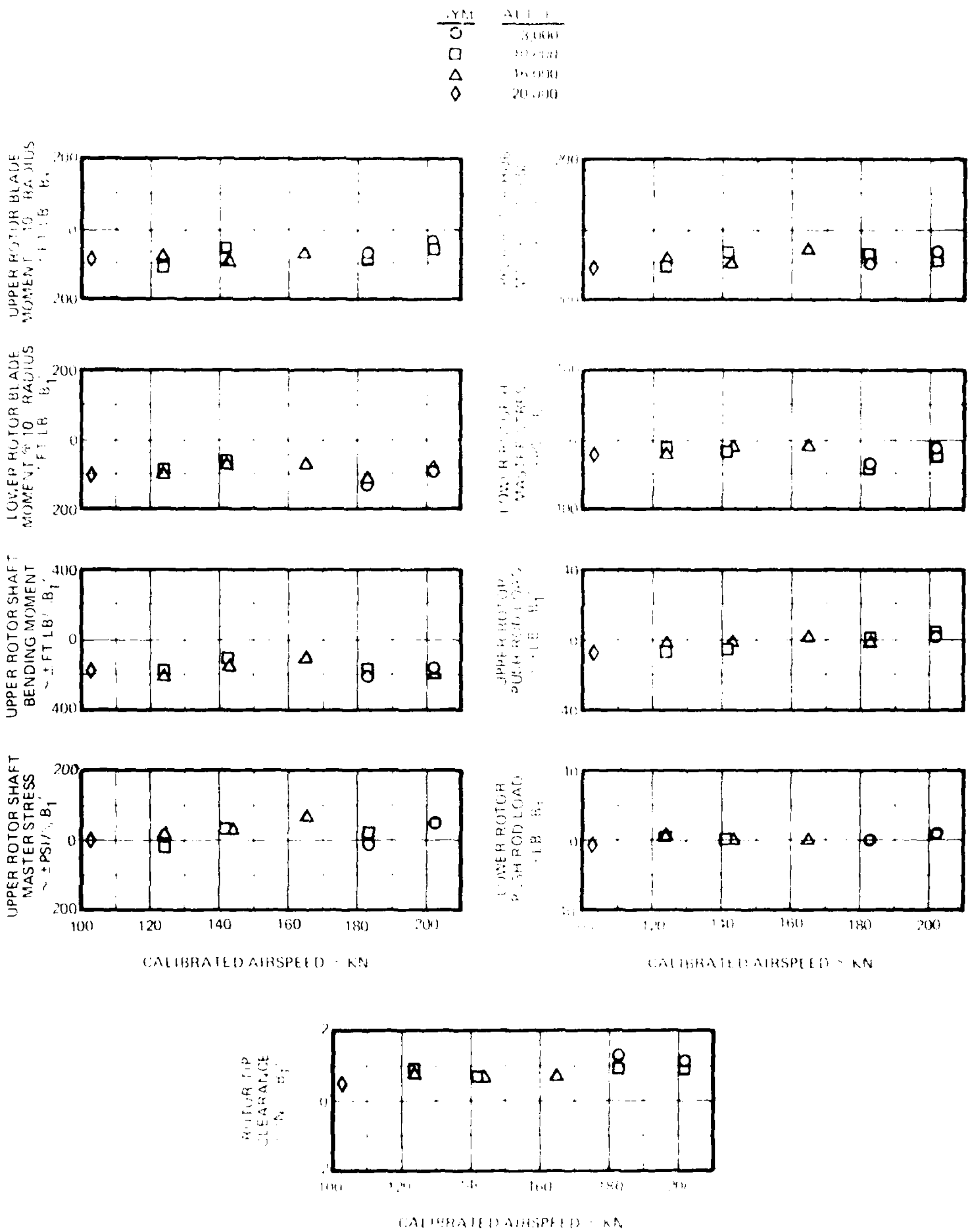

Figure 27. Effects of Differential Lateral Control on Rotor Component Loads and Stresses.

2. The B_1' derivative did vary slightly with airspeed.

3. The sign of the derivative for the rotor parameters was negative, indicating that increasing B_1' decreased the rotor load/stress. The exception to this was rotor tip clearance, which increased nominally 1 inch for every 10-percent increase in B_1'. This was expected, due to the decrease in rotor loads.

Effect Of Collective

Rotor collective pitch variations were flown at the same altitudes and flight conditions as the B_1' investigation presented in Figure 27. The collective pitch data matrices are presented in Figure 28. The data correlate very well with the previous data presented in Reference 2. The results of the test indicate the following:

1. The sensitivity of the rotor loads to collective pitch did not change with altitude.

2. The sign of the collective derivatives for most rotor parameters was positive, indicating that increasing collective increased the loads. This effect was opposite to the B_1' effect (Figure 27).

3. A comparison of the collective (Figure 28) and B_1' (Figure 27) sensitivities on rotor loads based on percent of control indicates that rotor load sensitivity to collective was approximately twice that of B_1'. However, the ratio of collective to B_1' measured in degrees of pitch on the advancing side of the rotor was 3.6:1. Therefore, in terms of degrees of control, the rotor loads are 1.6 times more sensitive to B_1' than collective control.

Effects Of Differential Longitudinal Control (A_1') and Differential Collective Control ($\Delta_{\theta t}$)

A matrix of differential longitudinal control and differential collective control was flown at 10,000 feet density altitude to determine the A_1' and $\Delta\theta_t$ effects on rotor loads. The differential longitudinal control was varied from 30 to 50 percent and the differential collective control from 40 to 60 percent at airspeeds of 160 and 180 KCAS.

Increasing positive differential longitudinal control increased the nose-up moment on the upper rotor and decreased the nose-up moment on the lower rotor. The net effect was opening the rotor tip clearance over the nose. However, the greatest portion of the rotor loads was not generated with

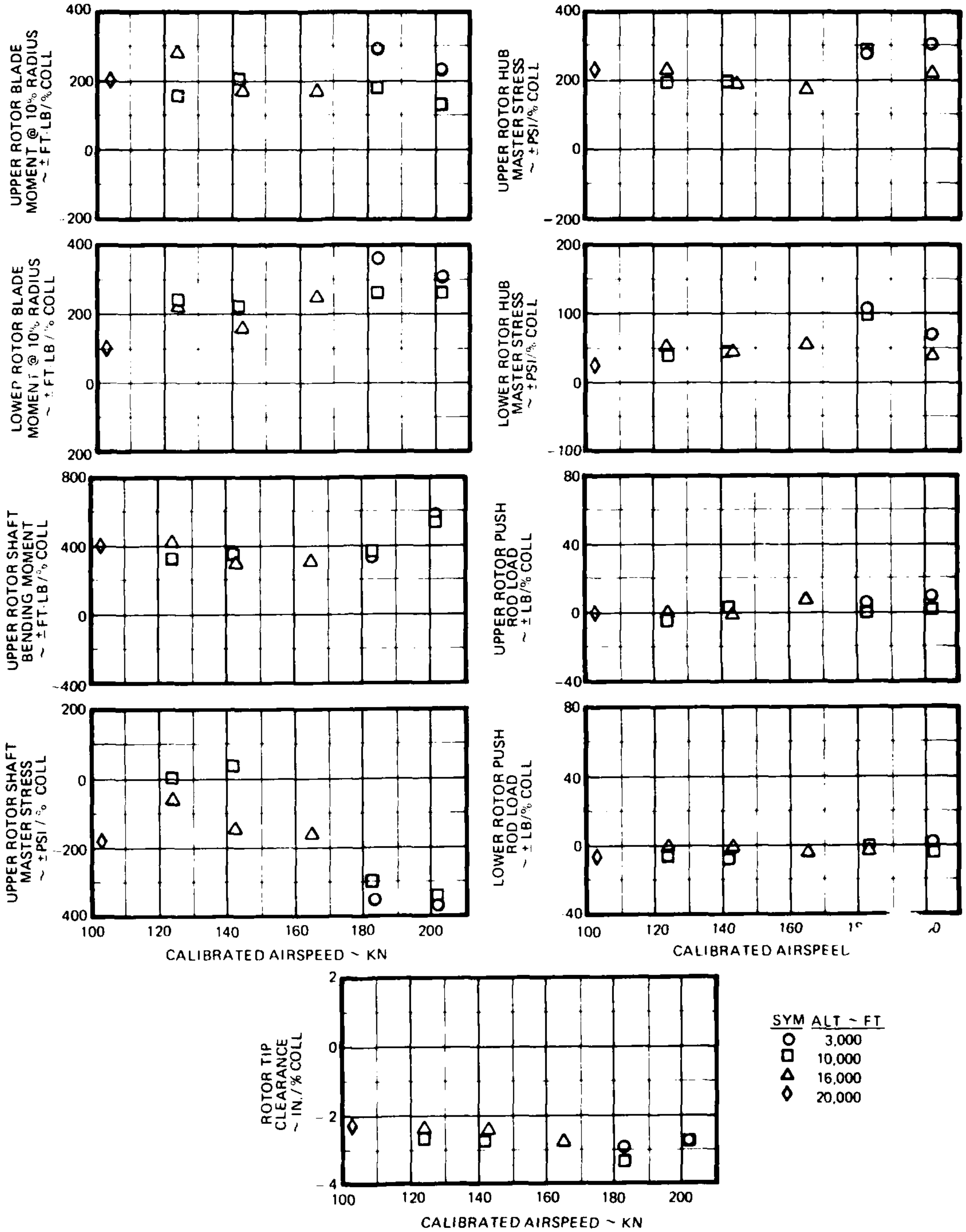

Figure 28. Effect of Collective Pitch on Rotor Component Loads and Stresses.

the blades in the fore/aft azimuth position. Therefore, A_1' had virtually no effect on rotor loads and a very minor effect on rotor tip clearance: + 0.04 in./percent A_1'.

Increasing differential collective trim increased the collective pitch on the upper rotor and decreased the collective pitch on the lower rotor. With the rotor in autorotation, the rotor loads were insensitive to change in $\Delta\theta_t$. Rotor tip clearance was the only exception, with a derivative of + 0.04 in./percent $\Delta\theta_t$.

Since the variations of A_1' and $\Delta\theta_t$ had virtually no effect on the rotor loads, no graphical data for those conditions are presented in this report.

Cockpit Control Schedule Development

It was indicated earlier in this section that a new cockpit control schedule would have to be considered if the altitude high-speed capability of the XH-59A was to be evaluated. The upper rotor hub master stress was the critical rotor structural parameter for the XH-59A aircraft. Qualitatively, it had been evident that the rotor loads were reduced when the rotor was in autorotation. The decision was made to explore the B_1'/collective influence with the rotor in autorotation.

Figure 29a is a typical presentation of the upper rotor hub master stress with B_1'/collective variations and the resulting stress relationship to the endurance values. This information was used to construct Figure 29b, which presents the B_1'/collective slope that would maintain the upper rotor hub stress at endurance. The upper rotor hub master stress was analyzed in this manner for all airspeed, altitude, and B_1'/ collective combinations tested. Analysis of this data indicated that the rotor loads were a function of calibrated airspeed and resulted in the family of collective/B_1' schedules of Figure 30.

Before establishing a final collective/B_1' schedule, the effect of the schedule on rotor speed was analyzed. Figure 29c is a typical illustration of the effect of B_1' on rotor speed and Figure 29d shows the effect of collective on rotor speed. A summary of the rotor speed derivatives with respect to B_1' and collective is presented in Figure 31. The rotor speed derivatives with collective were found to be a function of true airspeed. The influence of B_1' was constant for any altitude and airspeed, but the collective influence on rotor speed decreased as airspeed increased. Figure 32 presents the predicted rotor speed for the B_1' family of curves shown in Figure 30. The data indicated that the maximum rotor

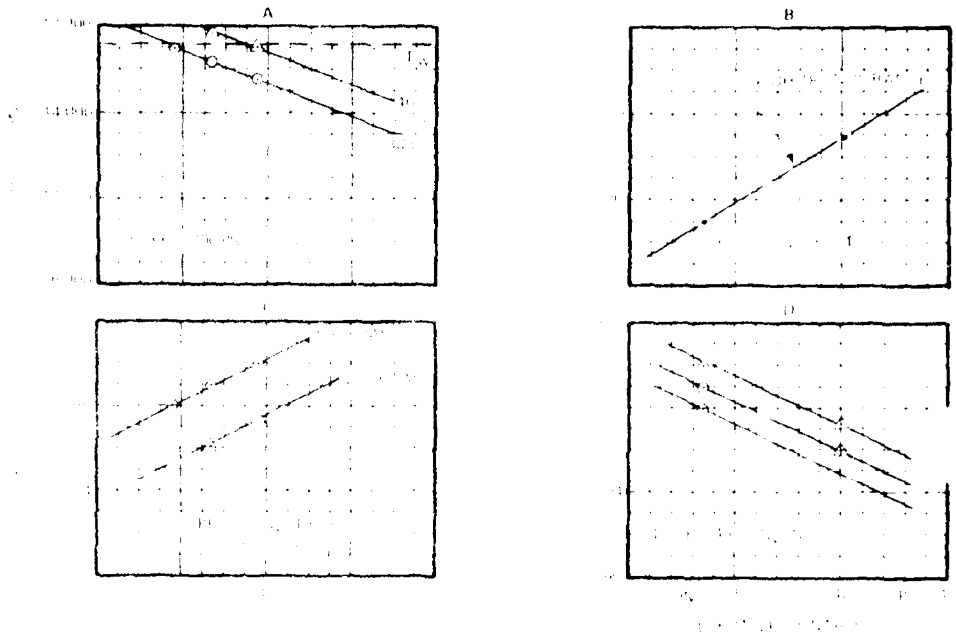

Figure 29. Effect of Control Variations on Upper Hub Stress.

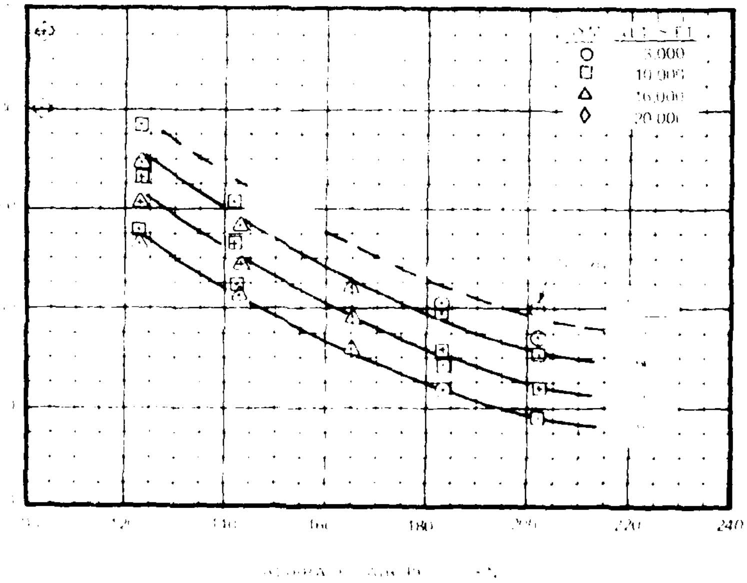

Figure 30. Collective/Differential Lateral Control Relationships.

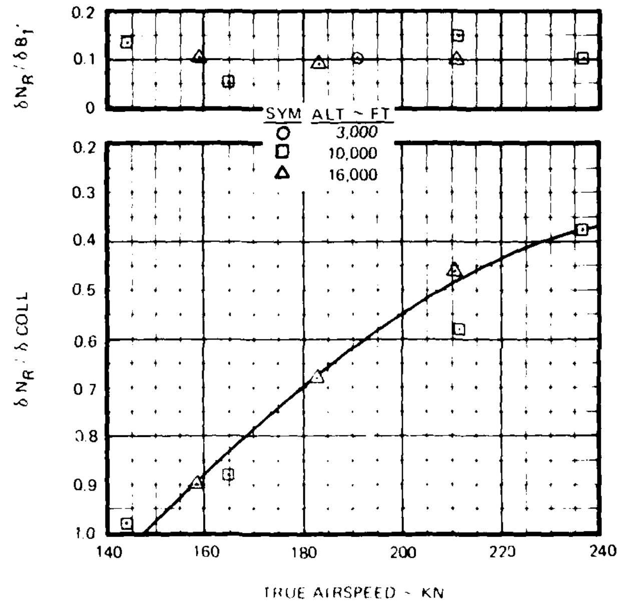

Figure 31. Effect of Collective and Differential Lateral Control on Rotor Speed.

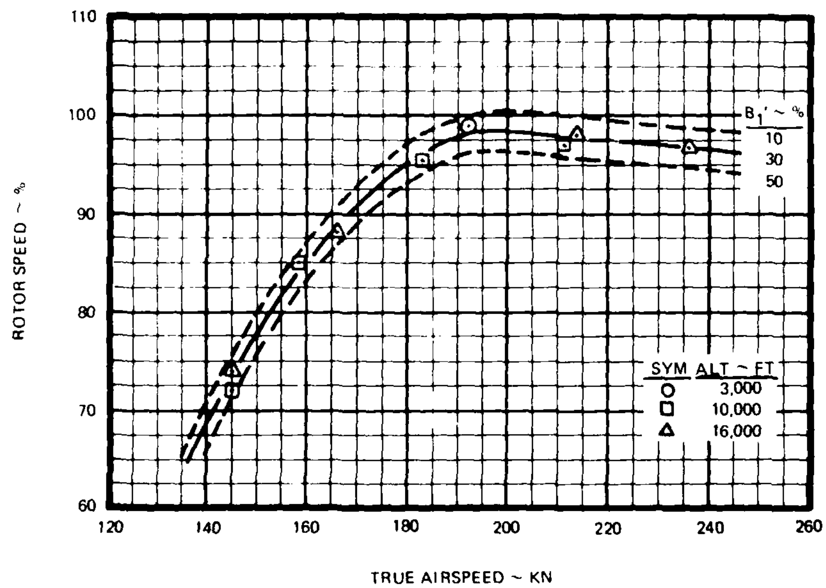

Figure 32. Rotor Speed Variation as a Function of Airspeed and Differential Lateral Control.

speed would be between 96 and 101 percent N_R at 195 KTAS. The rotor speed would then decrease as airspeed increased due to advancing tip Mach number. (It was apparent that at high airspeeds the XH-59A rotor system could not be operated at lower rotor speeds while maintaining the rotor stress within endurance.)

With the knowledge that collective/B_1' combinations were possible to allow high-speed flight and maintain rotor loads under endurance, a practical cockpit control schedule needed to be established. (It would be impractical to fly a constant B_1' and vary collective as indicated in Figure 30. Therefore, the collective/B_1' cockpit control schedule of Figure 33 was established.)

It must be emphasized that the procedure just outlined and the resulting cockpit control schedule established were required for the XH-59A rotor system. This procedure was mandated by the structural limits of the rotor system. A second generation rotor system with a stronger hub and modified blades would not require such a schedule.

Effects of Altitude

Figure 34 indicates the level-flight test conditions presented for Figures 35 and 36 and their relation to the recommended cockpit schedule. The data of Figure 35 demonstrate that the rotor loads for all trim level-flight conditions remained at or below endurance values throughout the airspeed range. The rate at which the rotor loads increase with airspeed with respect to the fixed cockpit controls is also evident.

The dive from 16,000 feet was flown with the collective on the recommended schedule; the B_1' was off the schedule but in the direction for reducing rotor loads. The rotor loads remained below endurance except for the upper rotor shaft master stress and the lower rotor blade bending moment. The high upper rotor shaft stress was indicative of a high rotor pitching moment. The fact that only the lower rotor blade bending increased appears to indicate that the increase in pitching moment was being produced by the lower rotor. The mechanism for this hypothesis is not understood. One possibility may be fuselage wake.

The other dive condition (10,000 feet) was flown at a higher collective setting. Higher collective decreases the stress on the upper rotor shaft (pitching moment) but increases the stress on the upper rotor hub. Therefore, it is evident that

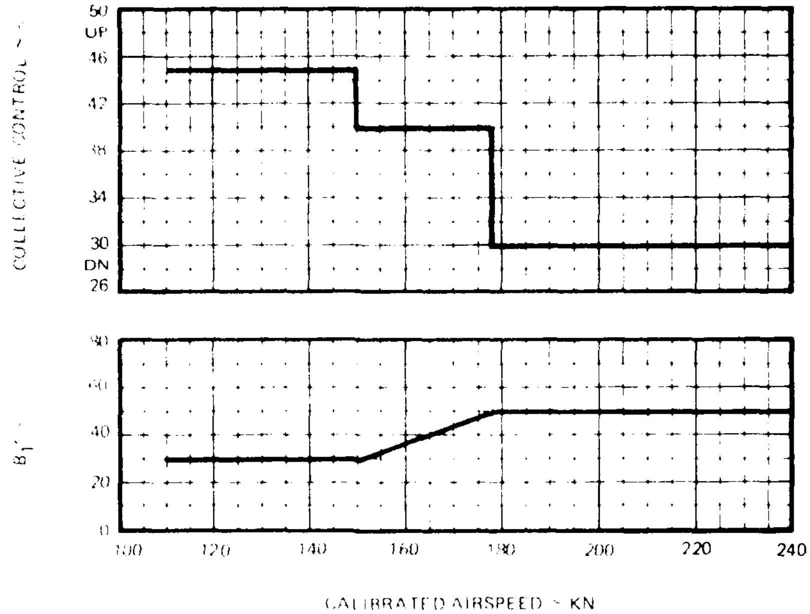

Figure 33. Cockpit Control Schedule.

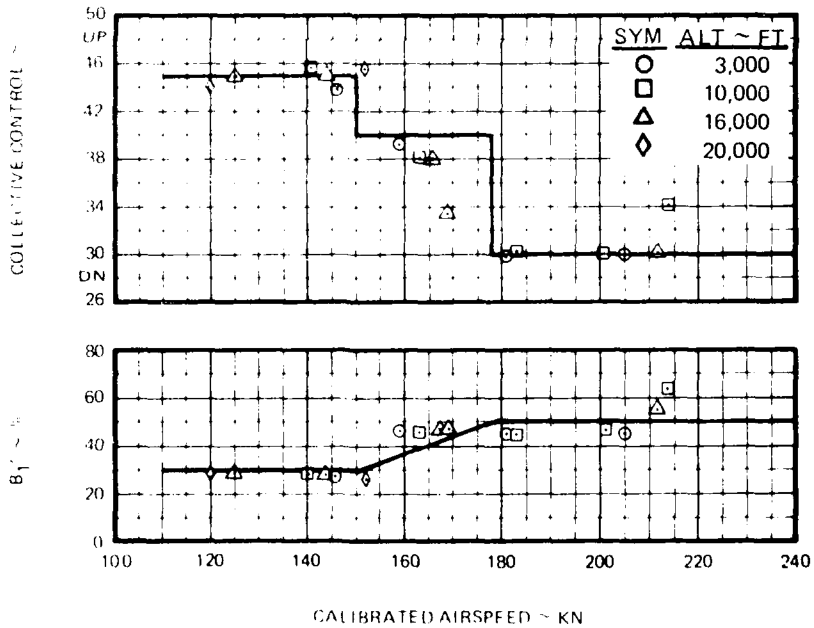

Figure 34. Cockpit Control Schedule Validation.

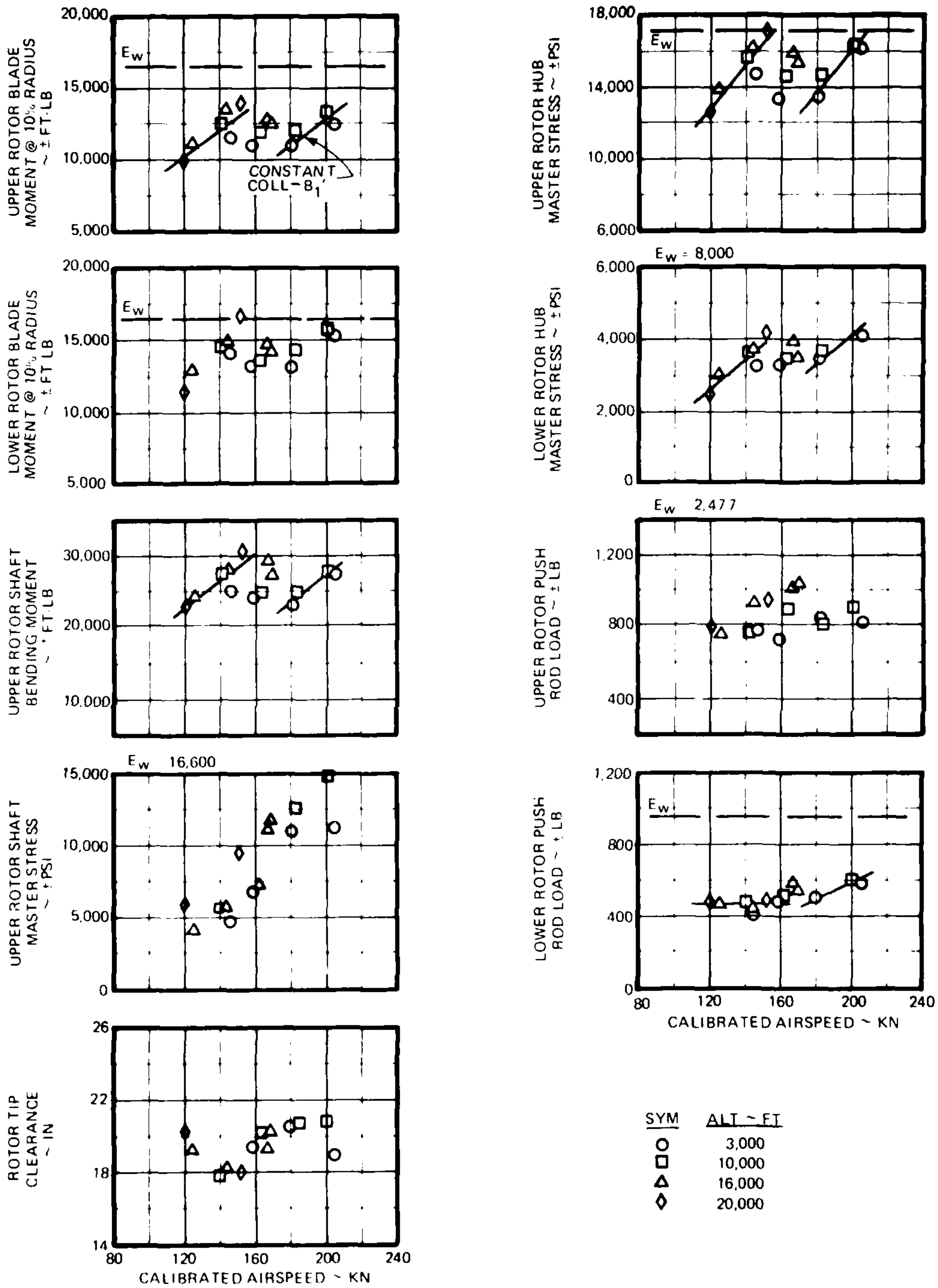

Figure 35. Rotor Structural Parameter Variation with Airspeed.

the XH-59A could not achieve higher airspeeds and still remain within the flight test guidelines without some other change to the aircraft. Reference 2 indicated the sensitivity of the rotor pitching moment to elevator angle. A change in elevator angle would permit higher forward speed. A change in elevator angle would be expected to permit higher forward speed.

Figure 36 presents the resulting rotor speed for the test conditions of Figure 34.

Effect of Load Factor

Several load factor maneuvers were flown at 10,000 and 16,000 feet density altitudes. The data are presented in Figures 37 and 38. A comparison of this data with Figures 191 and 192 of Reference 2 shows no change in rotor tip clearance or control loads with altitude. Analysis of the oscillograph waveforms indicates that at the high load factors the upper rotor push rods experience a small increase in the 7th harmonic. The lower rotor push rods experience a small increase in the 11th harmonic. These may be indications of incipient stall; however, classical rotor stall was not encountered for the conditions presented.

Effect of Roll Rate

Left and right roll rates of 30 degrees per second were achieved at 10,000, 16,000, and 20,000 feet density altitude. Figure 39 demonstrates that the rotor tip clearance during roll rate maneuver did not deteriorate with altitude. The tip clearance characteristics at altitude remained the same as those indicated at 3,000 feet in Figures 171 and 193 of Reference 2.

Effects of Sideslip

The scope of the test program did not allow for a detailed investigation of the effects of sideslip on the XH-59A with altitude. However, 8 degrees of sideslip left and right were flown at 180 knots at 10,000 feet for a comparison with the 3,000-foot data of Reference 2, Figures 194 and 195. The data of Figure 40 show no significant change to rotor structural behavior with sideslip at 10,000 feet. However, the angle-of-attack trend with sideslip at 10,000 feet was much flatter than at 3,000 feet. This shallow gradient of angle of attack and the slight reduction in rudder antiflutter overbalance weights resulted in less excitation and response of the rudder to rotor wake impingement in right sideslip at 10,000 feet. The response characteristics of the fuselage/tailcone attachment were not affected by altitude.

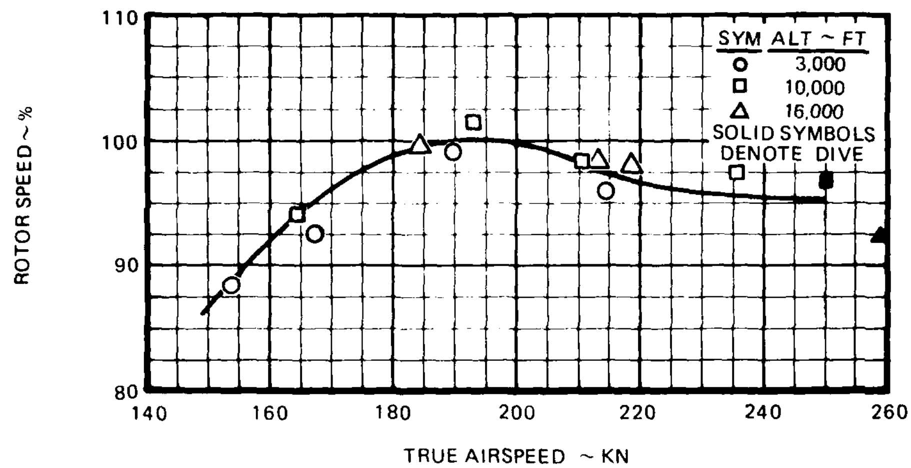

Figure 36. Rotor Speed Variation with Airspeed.

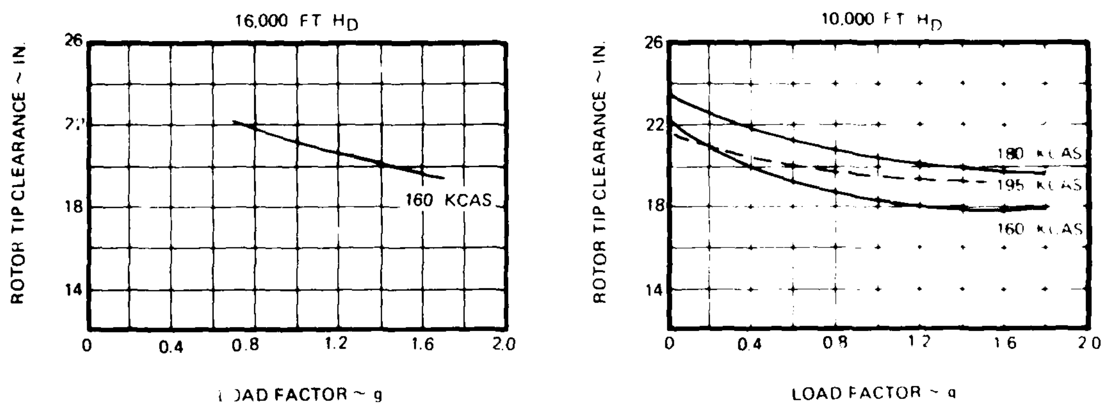

Figure 37. Effect of Load Factor on Rotor Tip Clearance.

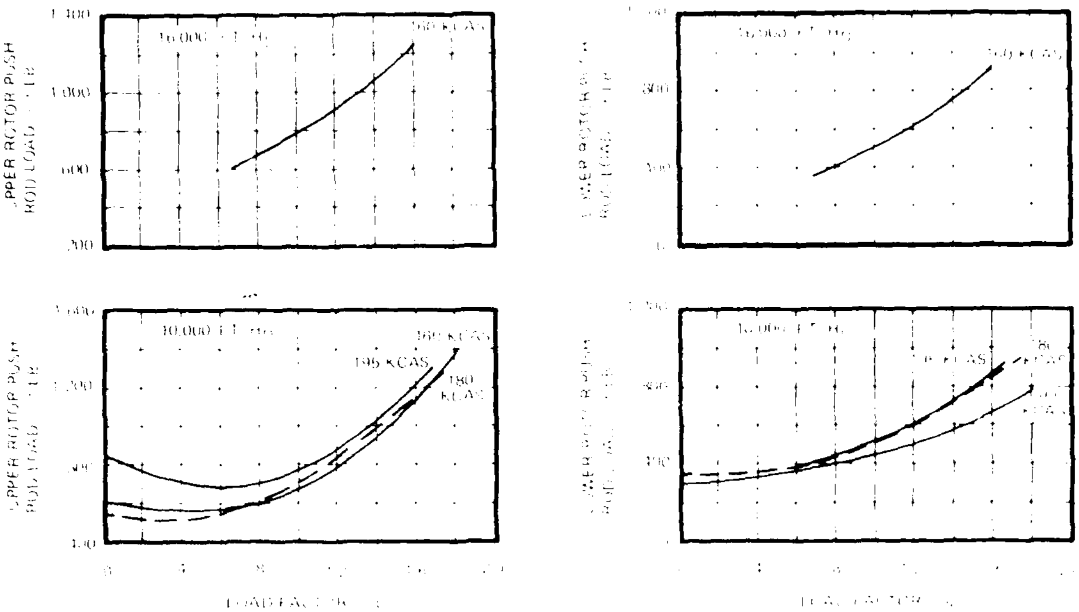

Figure 38. Effect of Load Factor on Control Loads.

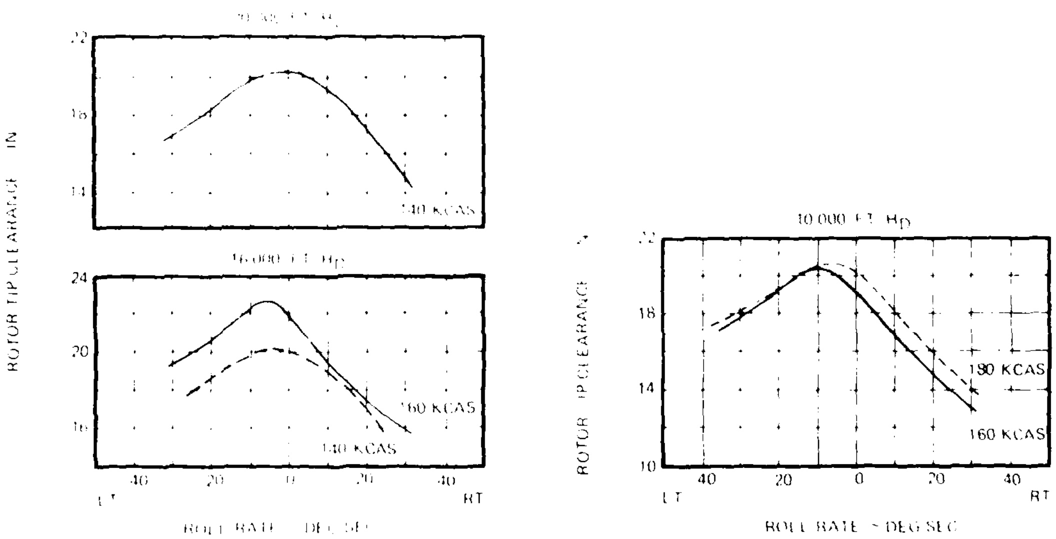

Figure 39. Effect of Roll Rate on Rotor Tip Clearance.

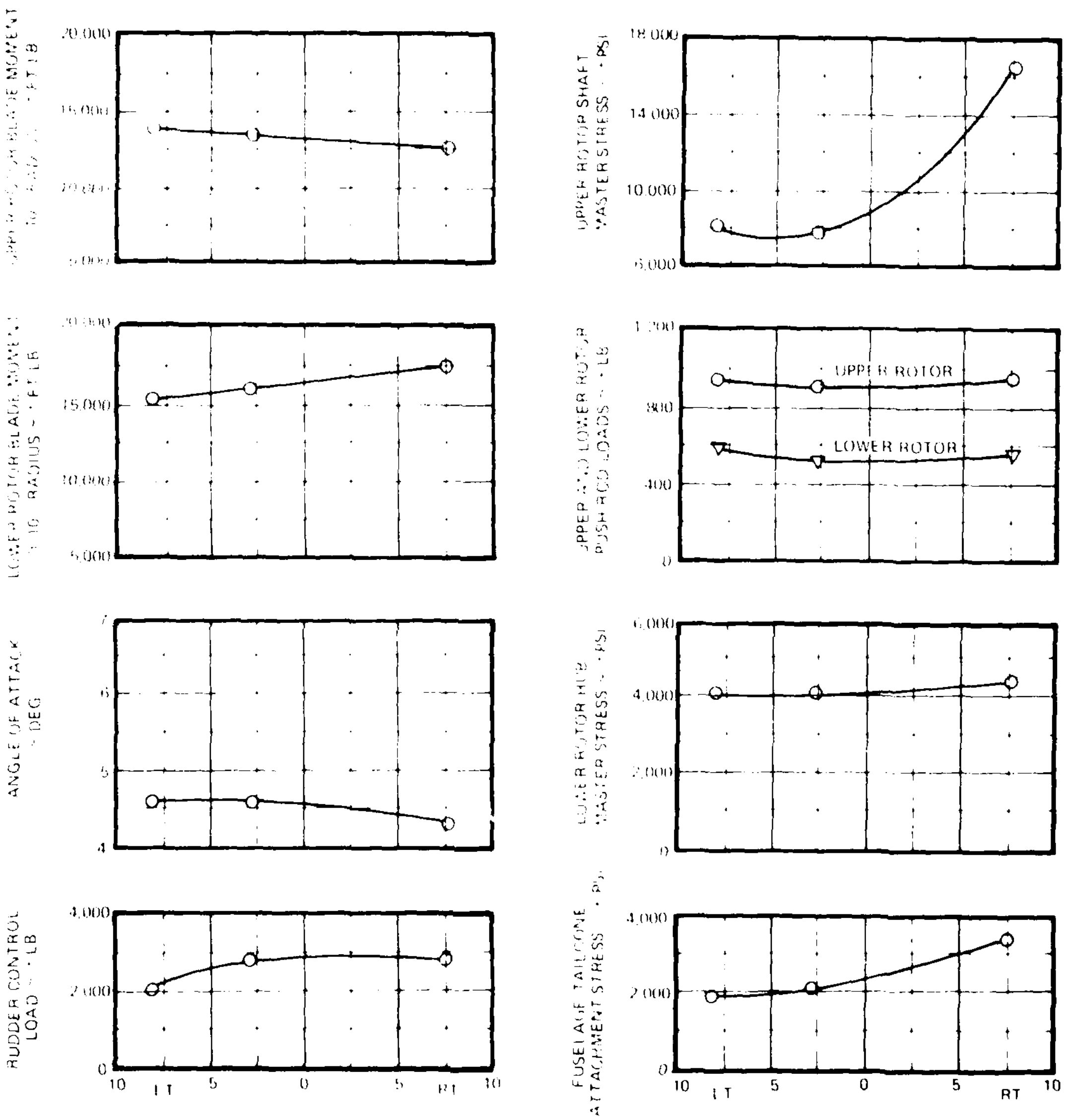

Figure 40. Rotor Structural Parameter Variation with Sideslip.

Center-of-Gravity Effects

The design limitations of the XH-59A airframe did not readily allow for evaluation of changes in aircraft center of gravity. In an attempt to acquire some information on the effects of center-of-gravity changes, the nose section of the aircraft was reinforced and a nose boom with increased wall thickness was installed. Provision was made for incremental increases in weight to be added to the boom to provide up to a 6-inch forward change in center of gravity.

The center-of-gravity testing was done at 3 and 6 inches. Flight conditions from hover to 210 knots were flown with a 3-inch forward center of gravity. The 6-inch forward center of gravity was only flown in a hover. There was not sufficient strength in the nose boom to allow a sufficient safety margin for reasonable avoidance or emergency maneuvering.

The test results are presented in Figure 41. The data show no change in rotor loads for forward flight conditions. However, the upper rotor shaft master stress reflects the change in rotor pitching moment required to hover.

Summary of Structural Results of the 0-Degree Crossover Auxiliary Propulsion Altitude and Center-of-Gravity Tests

1. The effect of differential lateral control (B_1') on rotor loads did not change with altitude.

2. The sensitivity of the rotor loads to collective pitch control did not change with altitude.

3. Differential longitudinal (A_1') and differential collective ($\Delta\theta_t$) changes did not affect rotor loads at altitude.

4. A new cockpit control schedule was developed resulting in the XH-59A flying more efficiently aerodynamically, while maintaining the rotor loads within endurance for all altitudes.

5. The effect of load factor on rotor tip clearance and rotor control loads did not change with altitude. Load factors were demonstrated to the predicted level for each test altitude.

6. Roll rates of 30 degrees per second were achieved at test altitudes up to 20,000 feet density altitude. The effect of roll rate on rotor tip clearance did not change with altitude.

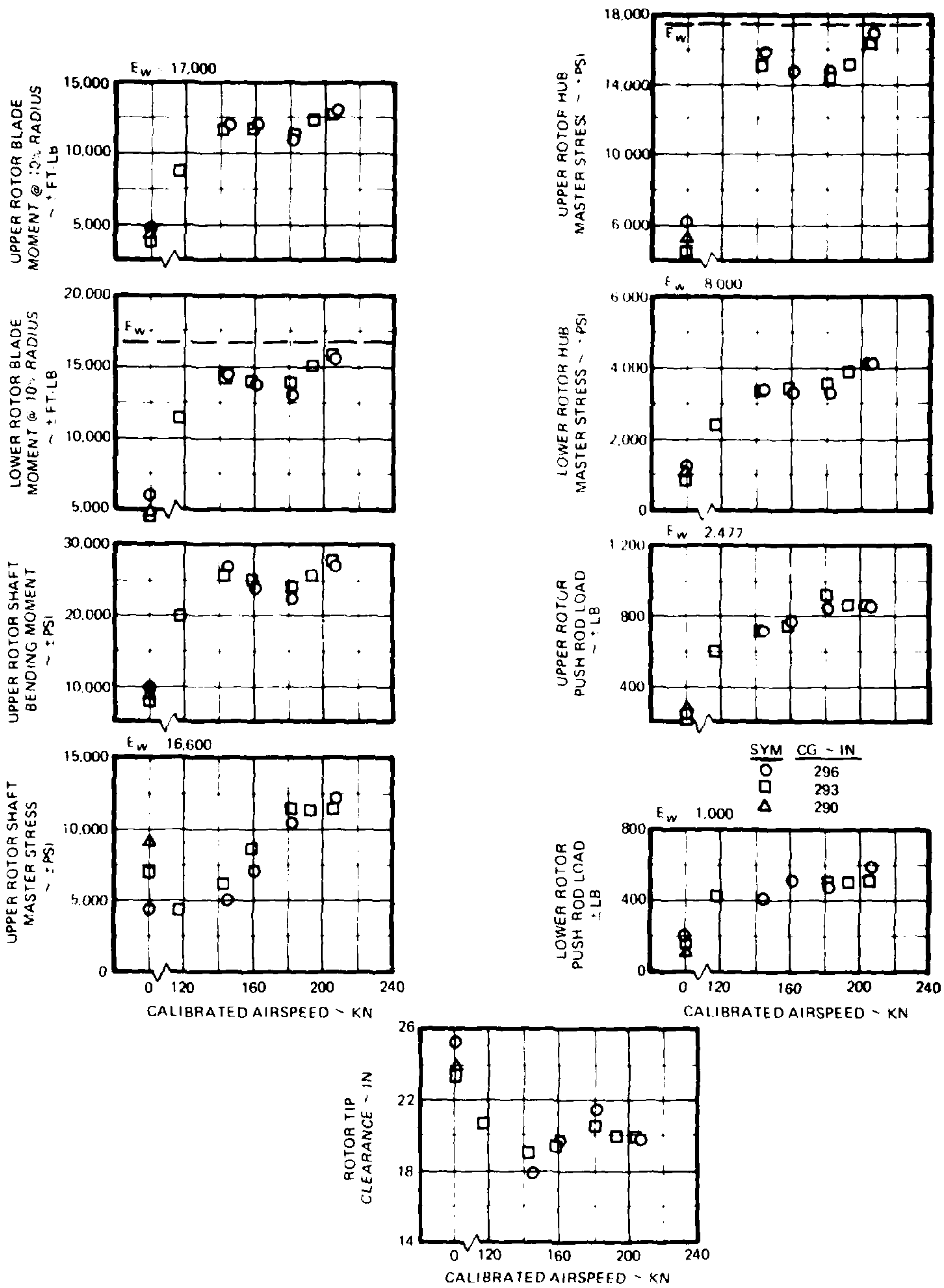

Figure 41. Effect of Center-of-Gravity Location on Rotor Component Loads and Stresses.

7. The limited sideslip data at 10,000 feet indicate no change of rotor loads with altitude. However, the angle of attack was reduced in right sideslip accompanied by a reduction in rudder rod load. This may substantiate the hypothesis that at 3,000 feet the rudders were in a rotor wake.

8. A 3-inch forward change in aircraft center of gravity did not change rotor loads in forward flight. Changes in center of gravity were evident in the upper rotor shaft stress for the hover condition.

DYNAMICS

Aircraft Aeroelastic Stability

The same monitoring and tracking procedures used for previous test phases were employed for the altitude test phase. The rotor was excited to check edgewise damping during envelope expansion by the pilot inputting a lateral cyclic doublet. The only significant mode of rotor response was a regressive in-plane blade mode of each rotor at a frequency of 1.4P in the rotating system. This mode was unchanged throughout the flight envelope and involved almost pure edgewise response.

Based on previous in-flight data analysis, minimum damping was experienced in autorotations in the helicopter mode. During this phase of testing, the rotors were flown in autorotation in most flight regimes. The damping checks were performed in 20-knot airspeed increments using the lowest anticipated collective setting. This level-flight condition yielded the lowest value of edgewise rotor damping. Figure 42 presents the rotor damping data for all test altitudes. The minimum damping observed occurred in the upper rotor at 140 KCAS and was above the 0.5 percent critical inherent structural damping measured in the blades. There was no discernable trending of rotor edgewise damping with altitude.

Upper rotor edgewise damping varied from 0.87 percent to 1.47 percent; lower rotor damping varied from 1.0 percent to 1.75 percent. Both rotors showed increased edgewise damping with increasing airspeed above 140 KCAS independent of altitude. There were no indications of elevator or rudder control surface instabilities (flutter) throughout the flight envelope. Analysis had shown the elevator to be flutter-free to over 600 knots in its locked configuration, while the predicted flutter-free speed of the rudders with their 60 inch-pound overbalance weights was in excess of 300 knots. A spectral analysis was performed on the elevator and rudder rod loads subsequent to each flight to verify that there was no buildup in the flutter frequencies.

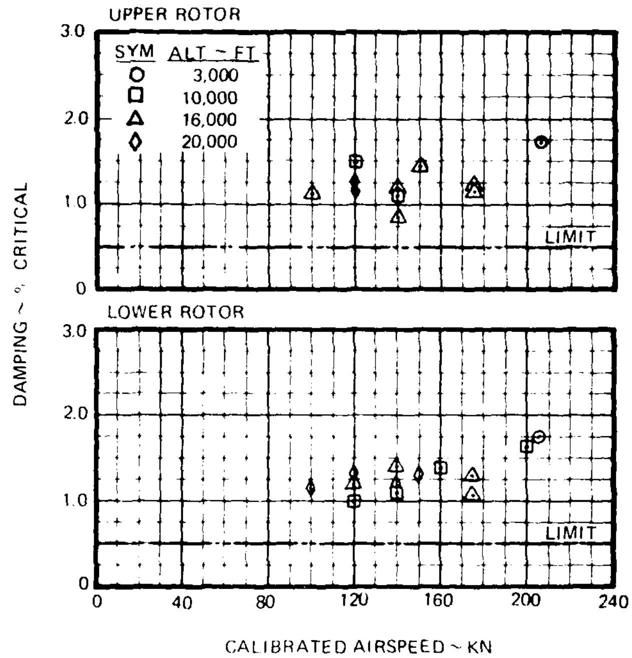

Figure 42. Blade Edgewise Damping.

Vibration

The source of airframe 3P vibration (which is the predominant frequency) is 2P rotor loads. Figure 43 shows these 2/rev loads as derived from harmonic analysis of the blade flatwise and edgewise moments. The edgewise contribution is relatively small in relation to flatwise and is independent of speed. The flatwise moments as predicted are increasing as a function of the square of the advance ratio (μ^2). A 20-percent reduction of the flatwise load is seen with increasing altitude from 3,000 to 16,000 feet.

The actual 3P vibratory load at any point in the aircraft is a function of both the 2P loads and the airframe response. The airframe response is very sensitive to the excitation frequency (rotor speed) as shown by the aircraft shake test (Reference 3).

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3. HANGING SHAKE TEST OF AUXILIARY PROPULSION ABC, SER-69026, Part 2, Sikorsky Aircraft Division, United Technologies Corporation, Stratford, Connecticut, 29 August 1980.

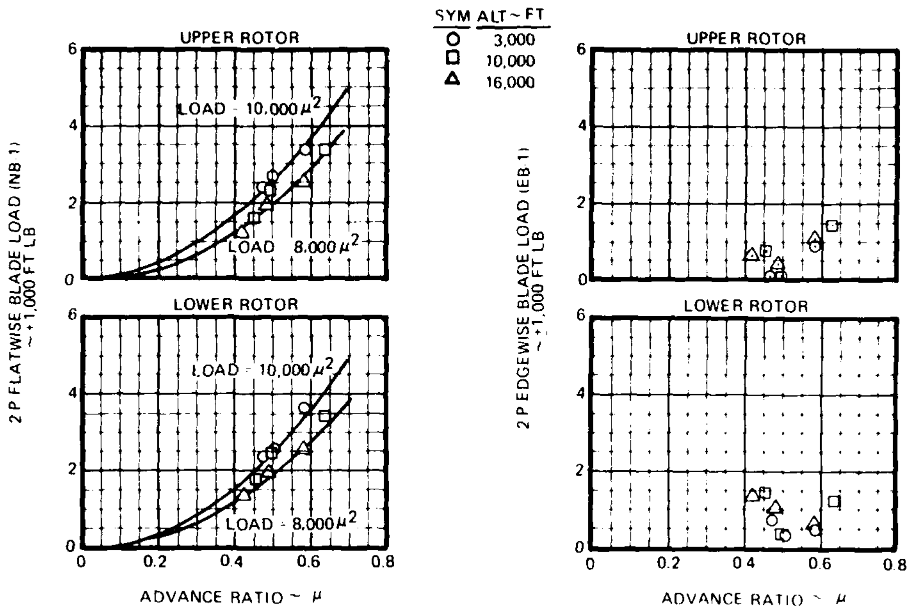

Figure 43. Rotor Loads.

With the test techniques employed (rotor flown in autorotation), maintaining a given rotor speed at the same airspeed with increasing altitude was not possible, nor was keeping a fixed rotor speed with increasing airspeed at a constant altitude. These rotor speed variations make evaluation of pure airspeed or altitude effects on cockpit vibrations difficult.

In an attempt to define the pure airspeed, rotor speed, and altitude effects, the 2P flatwise load and airframe vibration were assessed against the shake test results (Figure 44). This comparison is not fully valid, because the shake test responses were developed with a pure pitch moment input at the head and the 2P flatwise load contains forces and moments in all axes. In 0-degree crossover of the rotor, only the vertical, longitudinal, and pitch components should be transmitted to the airframe. The results of this comparison show that the shake test trends are generally substantiated. The fact that total cancellation of the lateral forces is not being achieved is also indicated in Figure 44.

Figures 45, 46, and 47 present the cockpit vibration trends for the two highest levels (copilot vertical and longitudinal) as a function of airspeed, rotor speed, and altitude. These trends were developed using the 2P flatwise blade loads

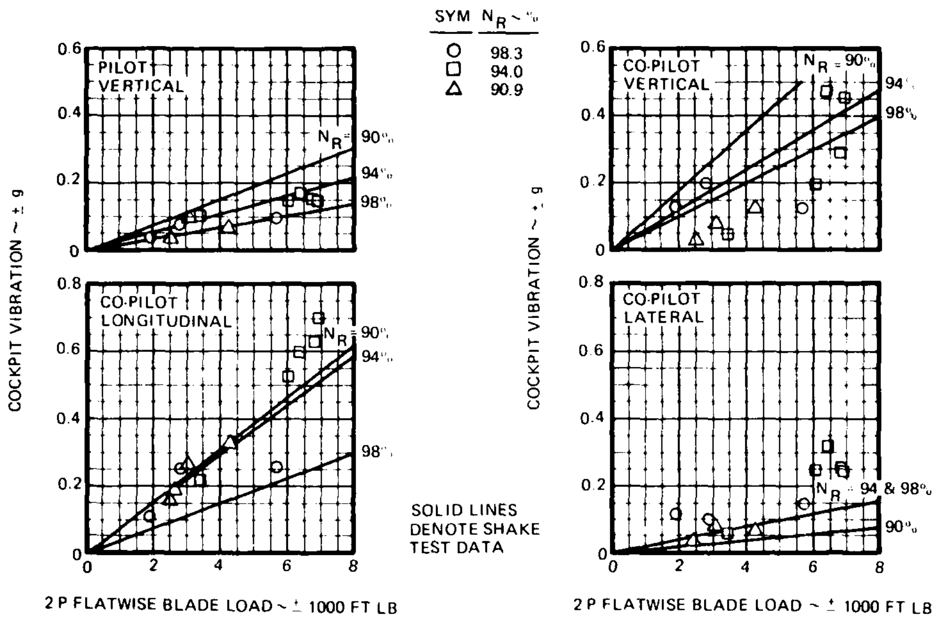

Figure 44. Comparison of Rotor Flatwise Loads and Airframe Vibration.

and the shake test response. The effect of airframe response characteristics is very evident in Figure 45. Here the data are presented for two different rotor speeds (98 percent and 90 percent). In one case they show the effect of the change of rotor loads as a function of advance ratio (i.e., airframe frequency response was held at 98 percent, while rotor loads were derived from 90 percent N_R). In the other case, the combined effect of the change in rotor loads and airframe response is shown. As can be seen, the impact of the change in airframe response is equal to the change in rotor loads for vertical response and five times greater than the effect of rotor loads on longitudinal response.

Figures 46 and 47 show the effect of altitude on cockpit vibrations for two different rotor speeds (98 percent and 90 percent). There is a reduction in cockpit vibration due to the change in rotor loads with increasing altitude.

Two flights were flown with the aircraft center of gravity moved 3 inches forward. The effect of this center-of-gravity change on cockpit vibrations is shown in Figure 48. Although the change in aircraft center of gravity was relatively small, the cockpit vibration trends observed show no change in vertical vibration and a reduction in longitudinal vibration at the higher airspeeds.

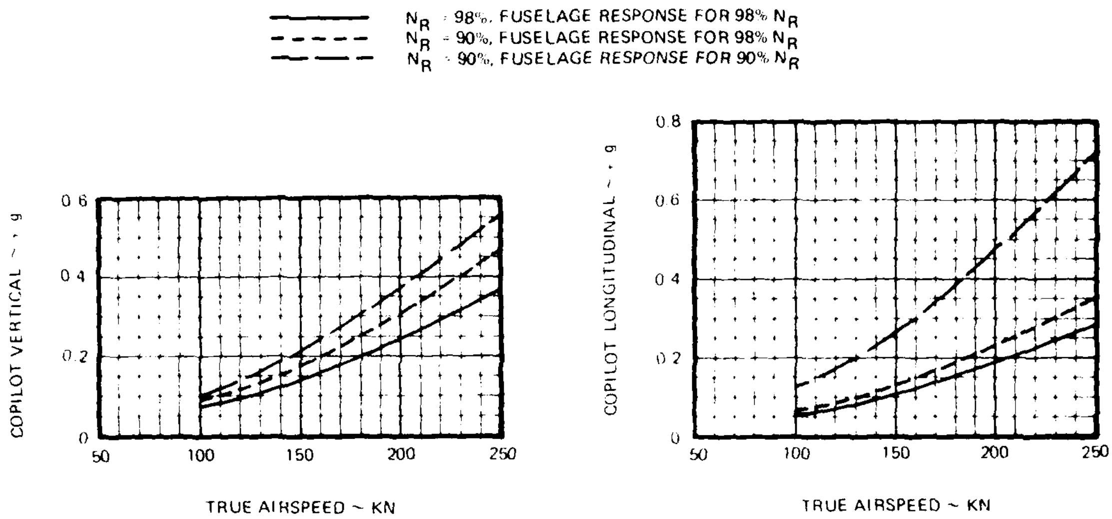

Figure 45. Effect of Airspeed and Rotor Speed on Cockpit Vibration.

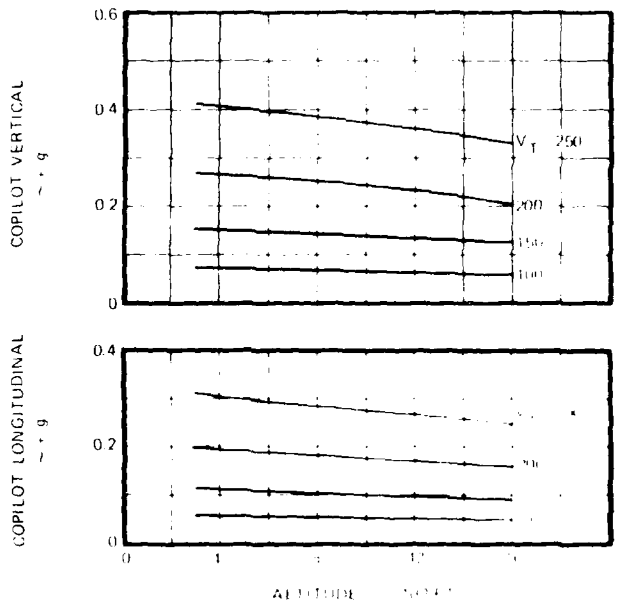

Figure 46. Effect of Altitude on Cockpit Vibration, Rotor Speed = 98 Percent.

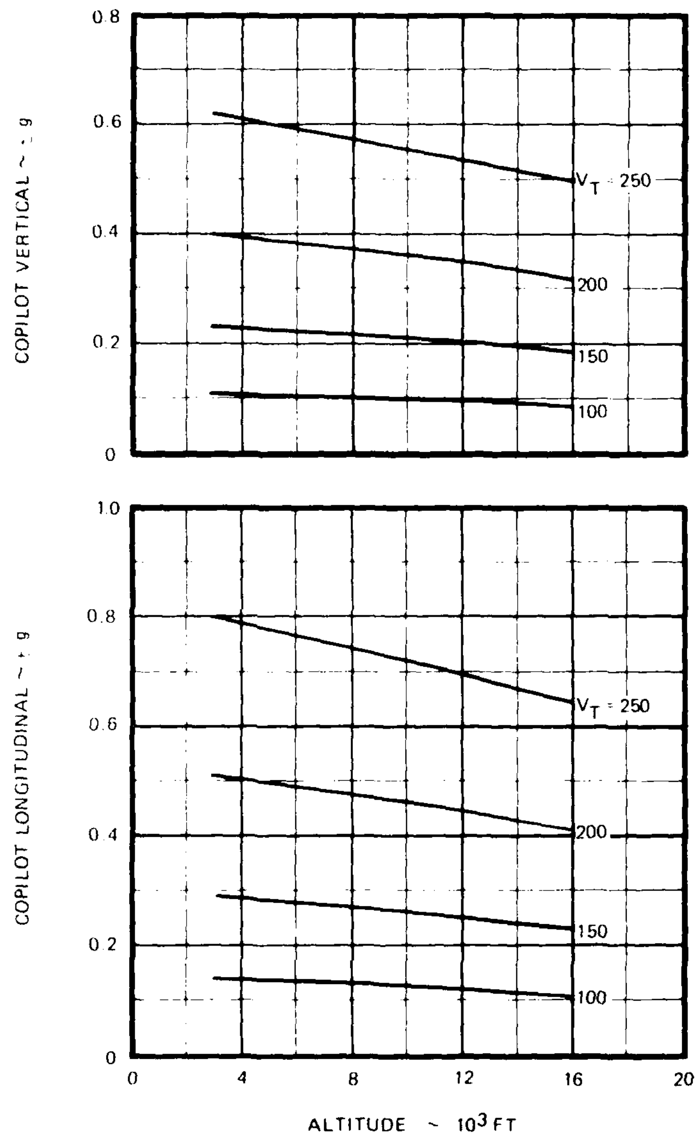

Figure 47. Effect of Altitude on Cockpit Vibration, Rotor Speed = 90 Percent.

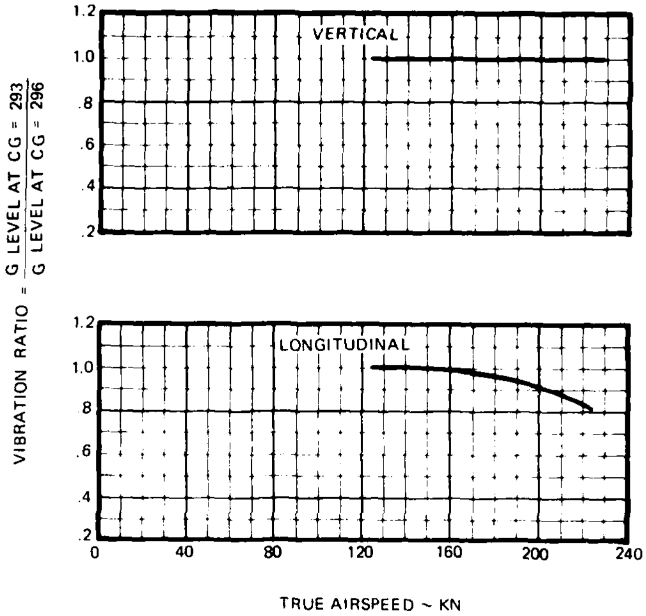

Figure 48. Effect of Center-of-Gravity Location on Cockpit Vibration.

OPERATIONAL TEST RESULTS

During May 1981 XH-59A flight tests in an operational environment were conducted in conjunction with the Army Aviation Development Test Activity. This program consisted of 8.0 hours of helicopter mode in the low-speed nap-of-the-earth (NOE) environment (Figure 49) and 4.6 hours of auxiliary propulsion mode in high- and low-speed contour flying (Figure 50). Demonstration of the full aircraft maneuver envelope at 2,000 and 10,000 feet was also included in this program. Figures 51 and 52 present time histories for typical hover and 40-knot NOE flight. Low- and high-speed low-level contour flight time histories are presented in Figures 53 through 56.

The aircraft performed well throughout the full NOE and contour environments. A final report has been published by the Aviation Development Test Activity reporting on this evaluation (see Reference 4). In general, the major achievements and findings during the program were:

NOE

1. Handling qualities excellent.
2. Excellent control power.
3. Compact rotor enhanced maneuvering in confined areas.
4. Downwash comparable to UH-60A.

Contour

1. Handling qualities excellent.
2. Excellent control power.
3. Maximum speed of 215 KTAS.
4. Acceleration capability outstanding.
5. Vibration levels, with no suppression equipment installed, were not detrimental to mission accomplishment.

Altitude

1. Maximum speed 240 KTAS.
2. Maneuver load factor 0 to 2.1 g.

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4. CW3 Leslie E. Scott, FINAL REPORT, SPECIAL STUDY OF THE XH-59A ADVANCING BLADE CONCEPT HELICOPTER, TECOM Report 4-A1-170-ABC-001, U. S. Army Aviation Development Test Activity, Fort Rucker, Alabama, August 1981.

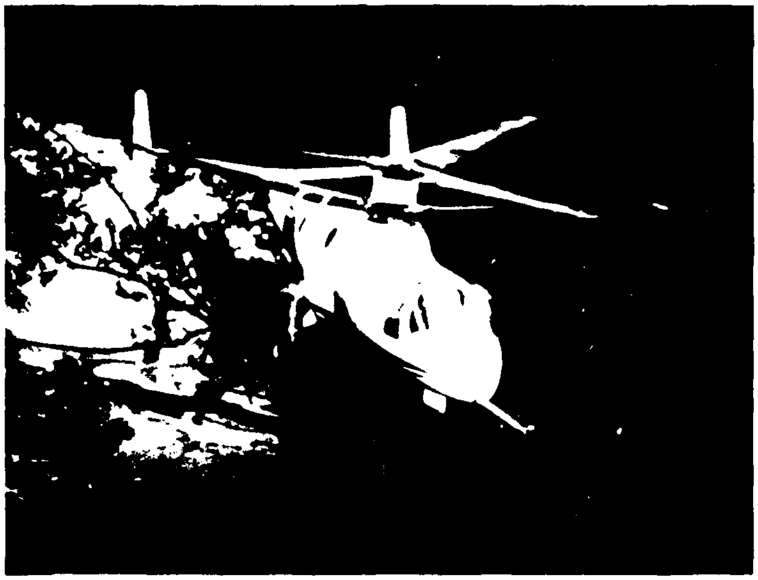

Figure 49. XH-59A in Nap-of-the-Earth Flight.

Figure 50. XH-59A in Contour Flight.

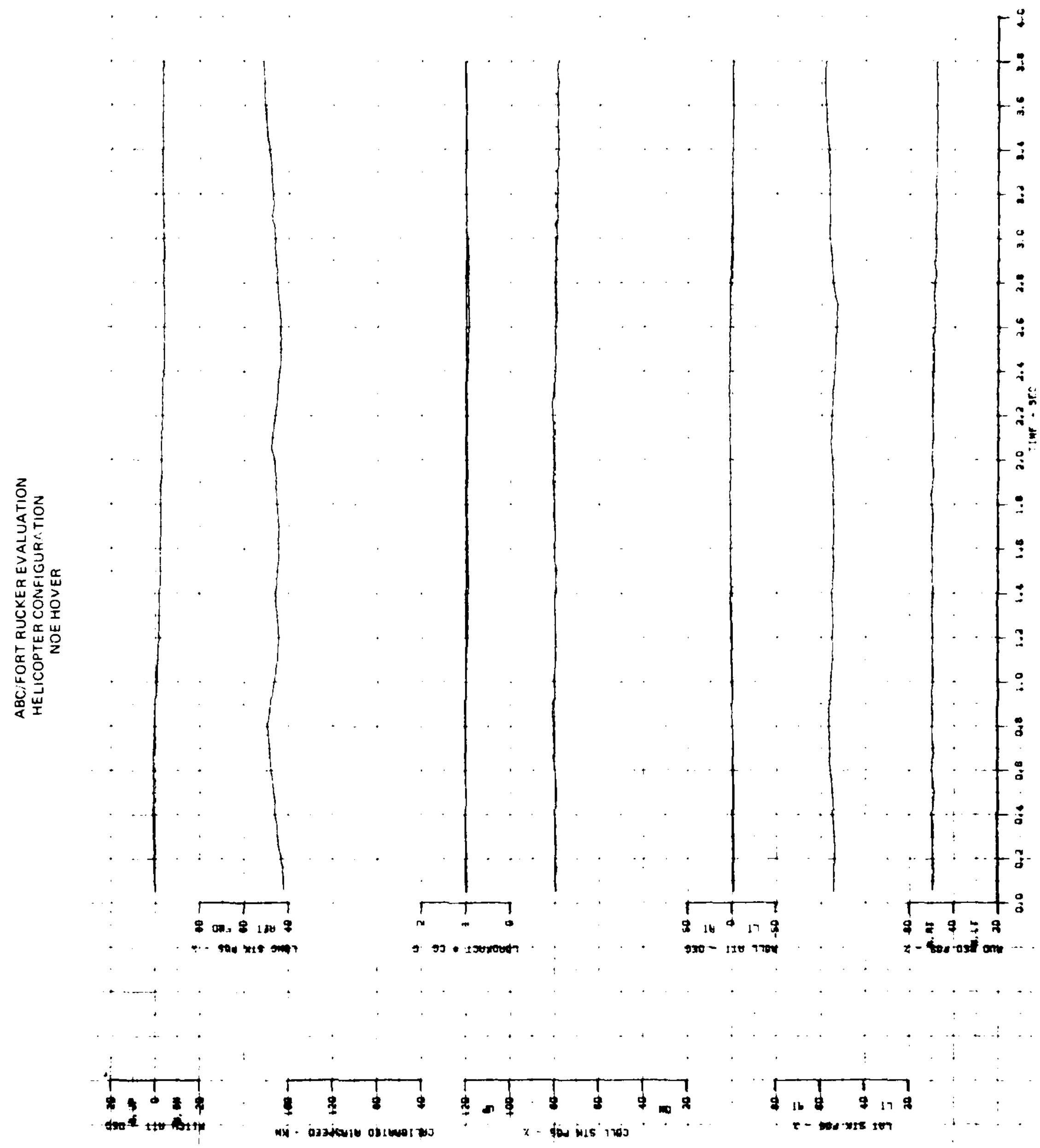

Figure 51. Time History, Hover NOE.

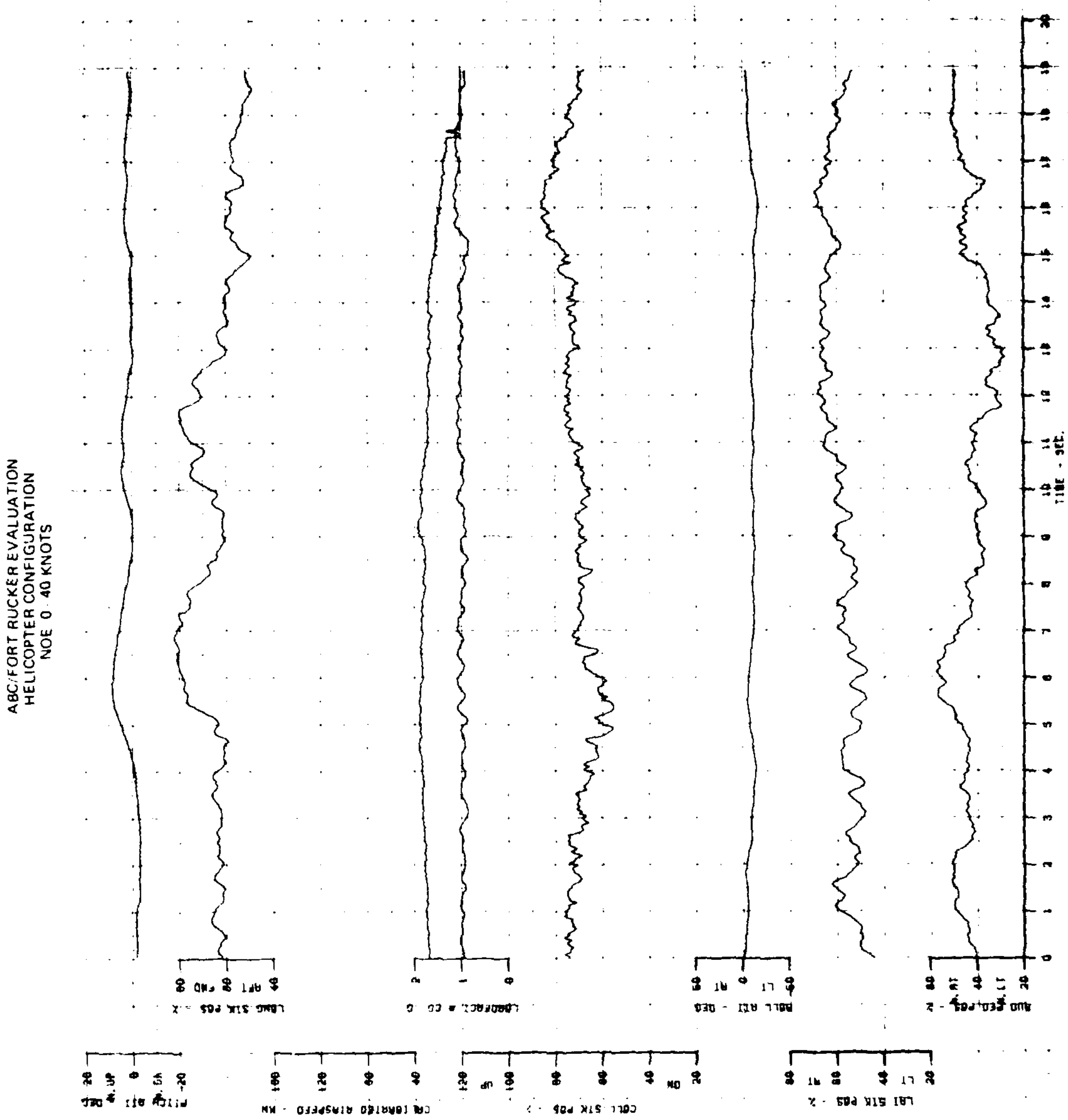

Figure 52. Time History, 40-Knots NOE.

ABC/FORT RUCKER EVALUATION
AUXILIARY PROPULSION CONFIGURATION
LOW ALTITUDE LOW-SPEED CONTOUR

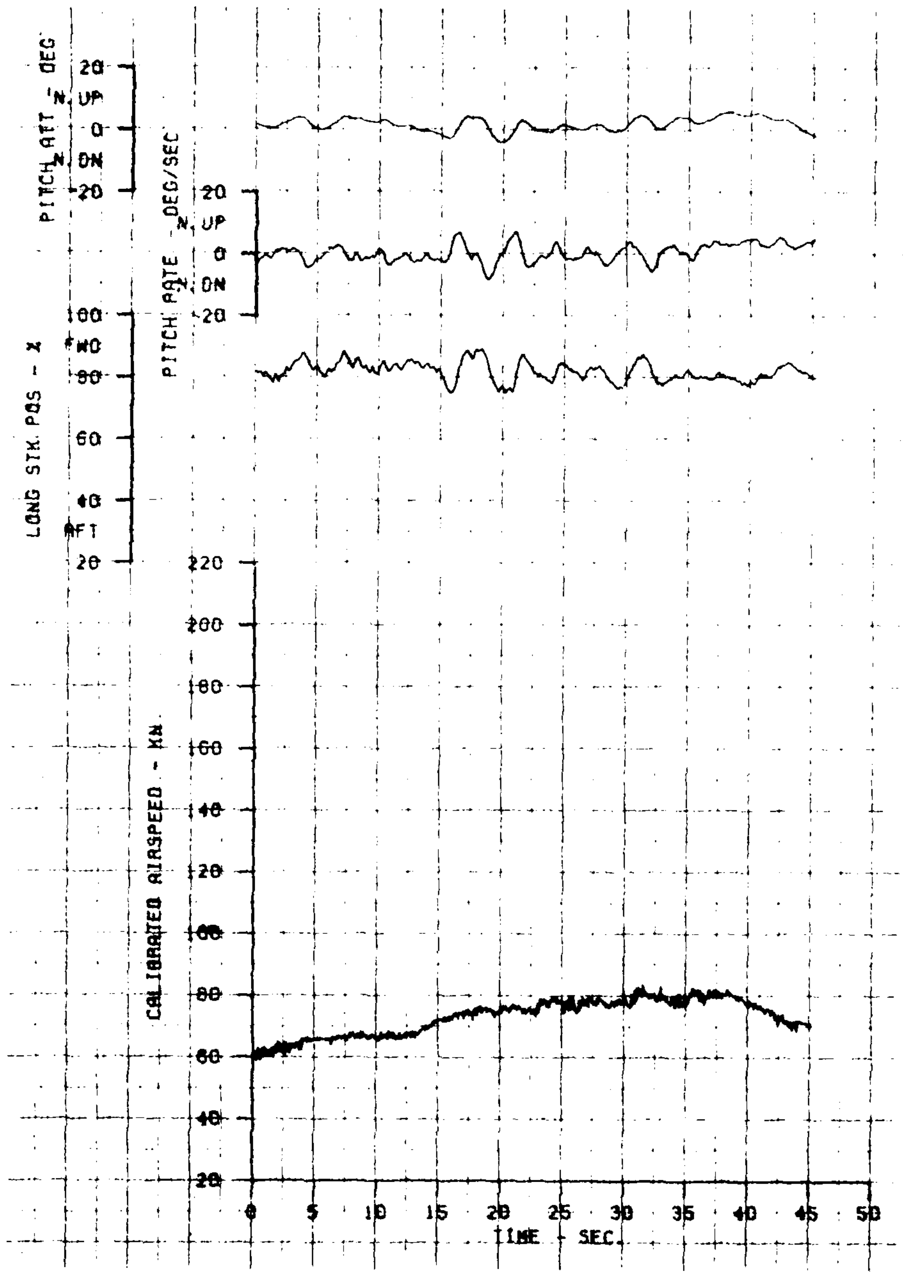

Figure 53. Time History, Low-Speed Contour.

ABC/FORT RUCKER EVALUATION
AUXILIARY PROPULSION CONFIGURATION
LOW ALTITUDE LOW-SPEED CONTOUR

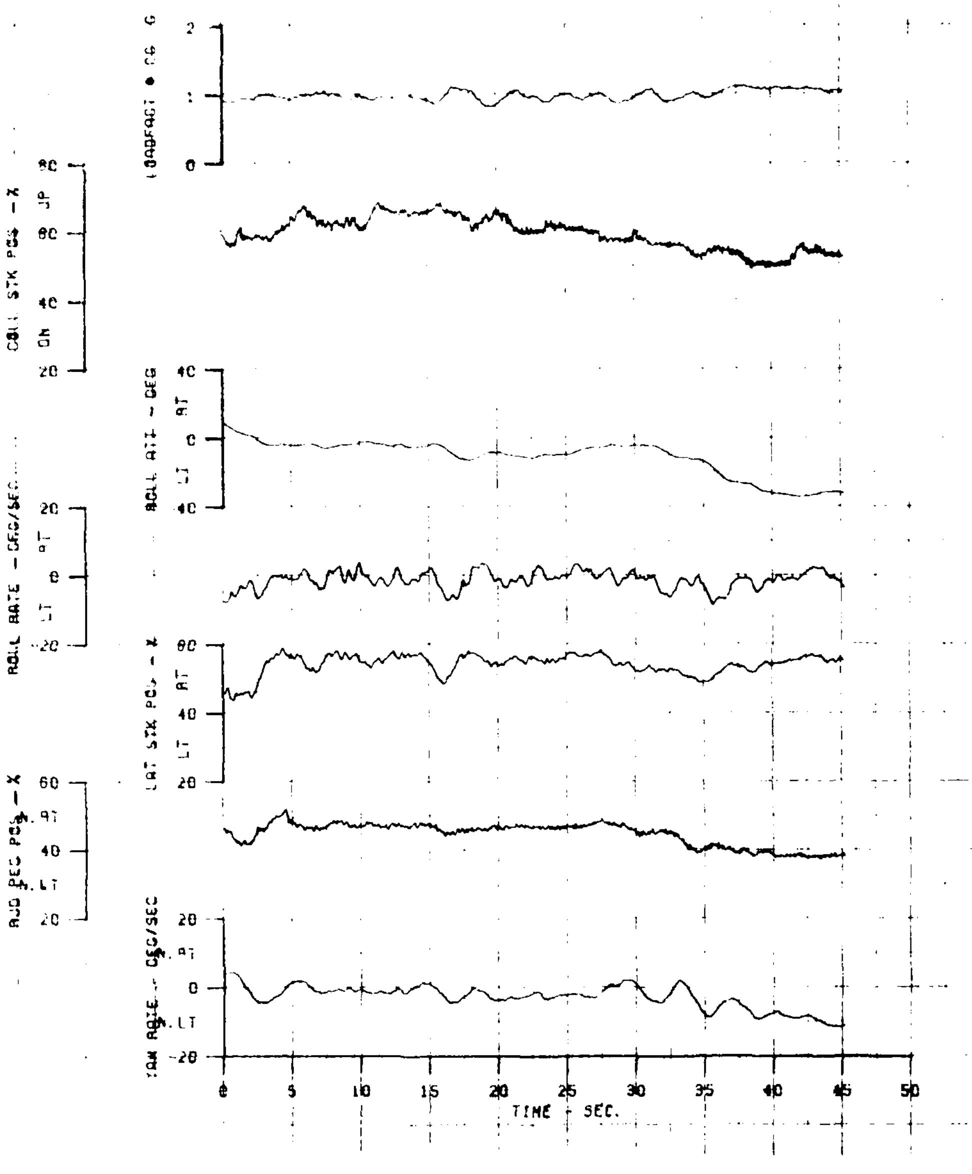

Figure 54. Time History, Low-Speed Contour.

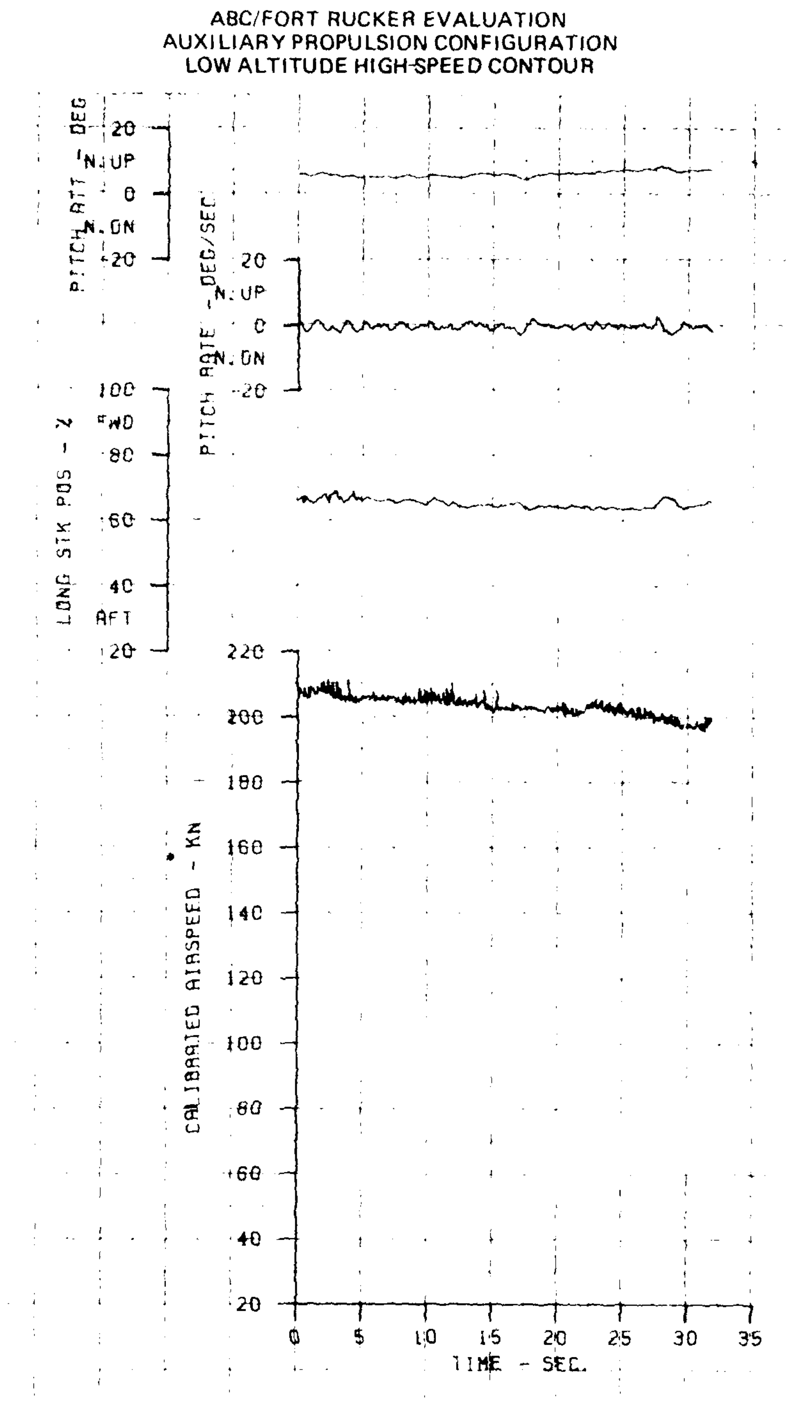

Figure 55. Time History, High-Speed Contour.

ABC/FORT RUCKER EVALUATION
AUXILIARY PROPULSION CONFIGURATION
LOW ALTITUDE HIGH-SPEED CONTOUR

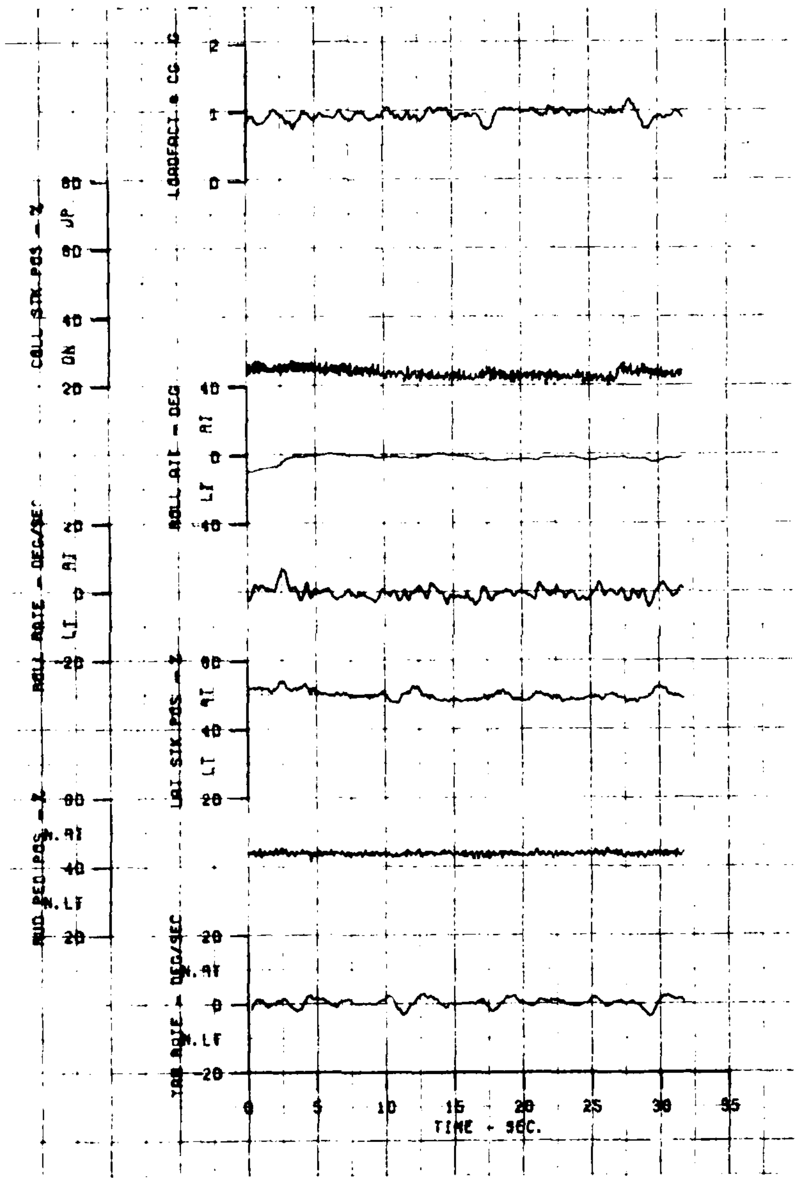

Figure 56. Time History, High-Speed Contour.

RESULTS AND CONCLUSIONS

The altitude and center-of-gravity envelope expansion and operational test programs, totaling 50 flight hours, support the following major results and conclusions:

1. The concept of developing lift primarily on the advancing blades of a rotor system to dramatically improve the lift potential has been proven. For example, at 180 knots the XH-59A demonstrated more than three times the lift coefficient of conventional helicopters. At higher speeds and during maneuvers greater ratios of lift increase have been shown.

2. Four goals were established for the altitude envelope expansion: altitude, speed, high-speed maneuver, and rotor speed reduction. Three were exceeded. Rotor speed reduction was demonstrated only at low speed.

3. Handling qualities have been excellent, with the aircraft stable in both the low-speed and high-speed flight regimes. The entire speed envelope has been demonstrated with SAS off. The high control power available has resulted in a very maneuverable aircraft and high damping precluded any tendency to overcontrol.

4. Tests conducted in an operational environment, in both low-speed NOE and high-speed contour flight, have shown the Advancing Blade Concept to be a viable alternative for future tactical aircraft.

5. Operational tests showed that, with no vibration suppression equipment installed, vibration levels were not detrimental to mission accomplishment.

6. The ability of the ABC rotor to maintain airspeed at altitude has been demonstrated.

7. The ability to achieve high altitude has been demonstrated.

8. The ability to maneuver at high speed and altitude has been demonstrated.

9. Forward flight lift/drag ratios predicted for this rotor system have been verified. Higher L/D values may have been attainable with rotor speed reduced more than was possible with the XH-59A rotor.

10. Rotor stress and control loads have been controllable as predicted and remained at acceptable values throughout the level-flight airspeed envelope.

11. Adequate tip clearance has been demonstrated throughout a significant maneuver envelope at all airspeeds and altitudes. Stiff blades and the absence of rotor stall result in very low control system loads.

12. Blade edgewise damping has remained stable; above minimum power required airspeed, the critical damping ratio is increasing.

13. Rate of climb and descent envelopes of 5000 ft/min have been demonstrated.

14. Excellent acceleration capability has been demonstrated, notably a level acceleration from hover to 185 knots in 28 seconds.

15. Initial flights to evaluate expanded longitudinal center of gravity showed no inherent problems.

RECOMMENDATIONS

Based on the results of flight testing to date, the following recommendations are made:

1. Operational analysis studies should be conducted by the Department of Defense to further define the advantages of speed in the operational environment. Potential ABC advantages of maneuverability, compactness, survivability/vulnerability, reliability/maintainability, handling qualities, radar cross section, and extended center-of-gravity range should be evaluated against potential missions.

2. Additional flight testing should be conducted with the XH-59A to investigate the following:

   - Further noise reductions in hover and intermediate speed cruise with lower main rotor tip speeds.
   - Reduction in rotor shaft stress (rotor pitching moment) with a controllable elevator.
   - Improvements in low-speed autorotative yaw control with inverted vertical tails.
   - Potential for reduced tail size and/or active elevator control.

3. A moving-base, real-time simulation of the ABC should be developed to further evaluate ABC design options and to compare the ABC to competing aircraft concepts.

4. Areas where the ABC analytical methods do not correlate with flight test results should be resolved.

5. Methods for reducing rotor system vibratory loads should be investigated. These could include higher harmonic control, which promises substantial vibration reduction with only a small penalty in aircraft weight.

6. A current-technology rotor and drive system should be installed to eliminate the present test restrictions of the XH-59A aircraft. A composite rotor system would allow an alternate value of blade twist and airfoil chord distribution to be investigated, while providing a substantially lighter rotor system. An integrated propulsion system with two engines powering both the rotors and an axiliary propulsion device would provide more power to the rotor, would be lighter and more fuel efficient, and would more closely represent the type of aircraft configurations proposed to meet future production requirements.

7. Subsequent to conversion to a twin-engine integrated propulsion system configuration, further investigations of rotor speed stability in maneuvering flight should be conducted.

8. Additional full-scale flight tests should be augmented by both full- and reduced-scale wind-tunnel tests, as appropriate. Model rotor testing and model tests of interactions of rotor/tail surfaces/pusher props or fans should be conducted.

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3. HANGING SHAKE TEST OF AUXILIARY PROPULSION ABC, SER-69026, Part 2, Sikorsky Aircraft Division, United Technologies Corporation, Stratford, Connecticut, 29 August 1980.

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LIST OF SYMBOLS

A_1	Longitudinal Cyclic Pitch, deg
A_1'	Differential Longitudinal Cyclic Pitch, deg
B_1	Lateral Cyclic Pitch, deg
B_1'	Differential Lateral Cyclic Pitch, deg
b	Number of Rotor Blades
C	Blade Chord, ft
C_T	Thrust Coefficient = $T/\rho \pi R^2 (\Omega R)^2$
C_T/σ	Thrust Coefficient/Solidity Ratio
E_w	Fatigue Endurance Limit
FPM	Feet/Minute, ft/min
H_D	Density Altitude, ft
KCAS	Knots Calibrated Airspeed
KN	Knots
KTAS	Knots True Airspeed
N_R	Rotor Speed, rpm, %
P	Rotor 1/Revolution Frequency
R	Rotor Radius, ft
SHP	Shaft Horsepower
$SHP/\delta \sqrt{\theta}$	Equivalent Shaft Horsepower
T	Rotor Thrust, lb
UH-18	Upper Hub Stress, Location #18
V	Velocity, kn
V_H	Power Limited Level Flight Speed
V_{KCAS}/δ	Referred Velocity
V_T	True Airspeed
W/δ	Referred Gross Weight

LIST OF SYMBOLS (continued)

α	Angle of Attack
Γ	Control Phase Angle
Δ	Delta, A Differential
δ	Pressure Ratio, P/P_o
ρ	Atmospheric Density, slugs/ft^3
θ	Collective Pitch, deg
$\sqrt{\theta}$	Square Root of Temperature Ratio, T/To
μ	Advance Ratio, KTAS/ΩR
σ	Rotor Solidity, bc/πR
ψ	Azimuth Position
ΩR	Rotor Tip Speed, ft/sec

Subscripts

f	Fuselage
L	Lower Rotor
o	Root Blade Angle
t	Trim
U	Upper Rotor
w.o.	Washout
.75	75-Percent Blade Station