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FLIGHT TEST OF THE AEROSPATIALE SA-342 HELICOPTER

Duane R. Simon, et al

Army Air Mobility Research and Development Laboratory
Fort Eustis, Virginia

August 1975

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EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
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The Eustis Directorate of the U. S. Army Air Mobility Research and Development Laboratory conducted an evaluation of the Aerospatiale Model SA-342 during November 1974. A total of 10 hours 20 minutes of flight time was accumulated during five test flights. Performance, stability, and control tests were performed to evaluate the increased capabilities of the SA-342 over the SA-341 upon which the design of the SA-342 is based. Both helicopters have fan-in-fin type antitorque control. The SA-342's handling qualities were generally very good, and it exhibited improved lateral-directional stability over the SA-341; however, trimming the aircraft directionally within the "sideslip deadband" was still difficult. The improved fan provided

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sufficient control moment to attain 50 knots in sideward flight. It is recommended that additional effort be expended to explore and resolve problems with the fan-in-fin design experienced both by Aerospatiale and by Sikorsky Aircraft.

PREFACE

The Aerospatiale Model SA-342 Gazelle helicopter was flight tested at the Aerospatiale flight facility in Marignane, France, during November 1974 by Mr. Duane R. Simon, Test Pilot, and Mr. Jimmie C. Savage, Flight Test Engineer, of the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory (USAAMRDL).

Special recognition is accorded to the following Aerospatiale personnel: Mr. J. Boulet, who took the test team into the French Alps for high-altitude tests; Mr. J. Henry, who demonstrated several of the helicopter's characteristics that were beyond the scope of these tests and provided communciations with the French controllers; Mr. J. Besse, who briefed the test team on the modifications that distinguish the Model SA-342 from the Model SA-341 and coordinated the test activities; and Mr. R. Dahan, who provided valuable assistance in reduction of the flight test data. Special recognition is also extended to Mr. D. Arents of the Eustis Directorate, USAAMRDL, who assisted in the preparation of this report.

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INTRODUCTION

BACKGROUND

1. The Aerospatiale Model SA-342 Gazelle helicopter was a derivative of the Model SA 341 Gazelle currently in use by the French and British Armies. The Model SA 342 was developed to integrate and demonstrate advancements in fan-in-fin helicopter technology which were developed to improve handling qualities and performance.

2. Flight test evaluation of the Model SA-342 Gazelle was conducted under a joint invitation from Kaman Aerospace Corporation, Bloomfield, Connecticut, and Aerospatiale, Marignane, France, to the U. S. Army Aviation Systems Command (AVSCOM).* AVSCOM subsequently tasked the Eustis Directorate, USAAMRDL to conduct the tests with one test pilot and one flight test engineer. The evaluation was performed at the Aerospatiale flight facility in Marignane, France.

TEST OBJECTIVE

3. The objective of the flight tests was to evaluate the improvements in handling qualities and performance provided by the modifications to the fan-in-fin antitorque control installed in the Model SA-342 helicopter. Results of the tests were to be compared to the test results obtained in previous Langley Directorate, USAAMRDL flight tests of the Model SA-341 helicopter conducted in July 1972.[1]

DESCRIPTION OF TEST AIRCRAFT

4. The Aerospatiale Model SA-342 helicopter flown during the tests was a modified SA-341. The SA-341 is a single-turbine-engine, single-rotor aircraft fitted with a fan-in-fin antitorque control. The fin is cambered to permit the fan to be unloaded in forward flight. Subsequent to the Langley Directorate evaluation, Aerospatiale made several modifications to the aircraft based upon both in-service operations and the Langley evaluation. These modifications were incorporated into a company-owned Model SA-341, making it a Model SA-342, and this was the aircraft tested for this report. Only one Model SA-342 helicopter has been produced to date. Major changes to the Model SA-341 which comprise the Model SA-342 are given in Table 1.

5. A general aircraft description and photographs are contained in Appendix A.

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*Letter, Kaman Aerospace Corporation, 11 October 1974, subject: Offer to AVSCOM of Opportunity To Conduct Flight Test Evaluation of the Aerospatiale SA-342 Gazelle Fan-in-Fin Helicopter.

[1] H. Kelley and T. C. West, *Flight Investigation of Effects of a Fan-in-Fin Yaw Control Concept on Helicopter Flying Quality Characteristics*, NASA TN D-7452, Langley Directorate, U. S. Army Air Mobility Research and Development Laboratory, Hampton, Virginia, April 1974.

TABLE 1. COMPARATIVE DESCRIPTIONS OF THE MODELS SA-341 AND SA-342 HELICOPTERS

	SA-341	SA-342
Engine	ASTAZOU IIIB, 590 hp (440 kw)*	ASTAZOU XIV, 858 hp (640 kw)*
Rotor Speed	378 rpm	387 rpm
Transmission Limit	493 hp (368 kw)	570 hp (426 kw)
Takeoff Gross Weight	3969 lb (1800 kg)	4189 lb (1900 kg)
Empty Weight	2000 lb (907 kg)	2026 lb (918 kg)
Useful Load	1969 lb (893 kg)	2163 lb (982 kg)
Tail Fan Design		
Airfoil Section	NACA 0016	63A2XX (thickness distribution unknown)
Blade Twist	-12° (root to tip)	-7° (root to tip)
Root End	not sealed	sealed
Pitch Range	-20° to 39°	-24° 49' to 40°
Fan Rotational Speed	5774 rpm	6000 rpm
	First Collective Detent: Designed to allow maximum rate input to prevent engine surge	Collective Detent: Indication to pilot of optimum performance for cruise flight; no prospect of surge with new engine
	Second Collective Detent: Normal power limit (approximately 2° below maximum available)	Collective Elastic Stop. Normal power limit (1.8° below maximum available)

*This does not reflect installation losses of approximately 27 hp (20 kw).

TEST SCOPE

6. The evaluation of the Model SA-342 helicopter was performed in a total of 10 hours 20 minutes of flight time, of which approximately 8 hours 30 minutes were productive. These hours were accumulated in five test flights at the Aerospatiale flight facility during the period from 18 November 1974 through 26 November 1974. The tests performed and the respective flight conditions are summarized in Table 2.

7. Installation, calibration, and maintenance of test instrumentation were performed by Aerospatiale. Test instrumentation consisted of a voice recorder and two 8-channel oscillograph recorders. The parameters recorded are listed in Appendix B. Support and assistance in data reduction were provided by the Aerospatiale Flight Test Department staff.

TABLE 2. FLIGHT TEST CONDITIONS

Test Condition	Gross Weight (lb)	Density Altitude (ft)	Temperature (°C)	Airspeed (kias)
Power Required for Hover	3750 to 4333 (1700 to 1965 kg)	IGE, -475 to 9050 (skid height 3 to 5 ft) OGE, -400 to -375 (skid height 75 to 100 ft)	-5 to 12	0
Power Required for Forward Flight	3500 avg (1587 kg)	1850	10	40 to 150
Power Required for Hovering Translations	"	IGE, -475 OGE, -400 to -375	6 to 12	-40 to +40 (longitudinal)* -50 to +50 (lateral)* -40 to +40 (quartering)*
Low-Speed Handling Characteristics	"	-300	12	" "
Controllability	"	300 to 2100	10 to 12	-40 to +40 (longitudinal)* -50 to +50 (lateral)* -40 to +40 (quartering)* 40 to 150 (forward)
Trim Change With Airspeed	"	1850	10	40 to 150
Trim Change With Vertical Speed	"	1850 to 5650	0 to 8	60, 90, 120
Static Lateral-Directional Stability	"	1850 to 5650	0 to 8	60, 90, 120
Maneuvering Stability	"	2750	7	0 to 150
Takeoff and Landing Characteristics	"	-850 to 9050	-5 to 12	0**
Boost-Off Characteristics	"	500	12	0 to 130
Autorotation	3000 (1360 kg)	-300 to 2750	7 to 12	0 to 90

*Paced Airspeeds
**Except for Autorotative Landing

TEST METHODOLOGY

8. Standard test methods[2,3] were used to acquire and evaluate handling qualities and performance data. These test methods are briefly described in the Results and Discussion section of this report. A Handling Qualities Rating Scale (HQRS), Appendix C, was used to augment pilot comments relative to handling qualities.

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[2] *Helicopter Stability and Control*, Flight Test Manual, Naval Air Test Center, USNTPS-FTM-No. 101, 10 June 1968.

[3] *Helicopter Performance*, Flight Test Manual, Naval Air Test Center, USNTPS-FTM-No. 102, 28 June 1968.

RESULTS AND DISCUSSION

GENERAL

9. Flight tests were performed on the Model SA-342 helicopter to evaluate performance and handling qualities during hover, translational flight, forward flight, autorotative flight, and maneuvering flight. Total power required, fan power required, and handling qualities were quantitatively and qualitatively evaluated. In brief summary: total power required for hovering turns in a 10-knot wind varied as much as 50 horsepower (37.3 kilowatts), depending on the direction of the wind; fan power required in hover was approximately 17 percent of total power required, with slightly less fan power required for OGE than for IGE hover; difficulty was experienced in trimming the aircraft within the sideslip deadband; directional damping was weak in hover; and directional instabilities were experienced while hovering in winds from the right. The Model SA-342 exhibited slightly improved static lateral-directional stability over the Model SA-341 as reported in reference 1, and the improved fan provided sufficient control moment to attain 50 knots in sideward flight.

PERFORMANCE

General

10. Minimal performance testing was conducted since the primary objective of the flight tests was to determine the handling qualities effects of configuration changes incorporated in the Model SA-342 helicopter. Engine torque was recorded, from which total power required was determined. Instrumentation for measuring fan thrust was not available, and fan drive shaft torque was recorded only during the last flight. Although it was desired to test at different rotor speeds, all tests were conducted at a constant rotor speed of 387 rpm and a fan speed of 6000 rpm due to the helicopter design characteristics. Hovering and low-speed power-required data were acquired at skid heights of 3 to 5 feet IGE and 75 to 100 feet OGE under the conditions listed in Table 2. Low-speed data were acquired by following a pace car equipped with an anemometer. Due to the time constraints on the evaluation, it was not possible to acquire performance data in calm wind conditions. High-altitude tests were performed during the one flight in which fan drive shaft torque instrumentation was installed, but due to uncertainty of wind direction and the rapidly changing wind velocities, the data were partially unusable. Pilot qualitative comments and some averaged quantitative data are discussed in paragraphs 17 and 36. Total power required for hovering turns in a 10-knot wind varied as much as 50 horsepower (37.3 kilowatts), depending on the direction of the wind. Fan power required in hover was approximately 17 percent of total power required, and slightly less fan power was required for OGE than for IGE hover. In forward flight at 140 KIAS, fan power required was less than 5 horsepower (3.7 kilowatts).

Hover and Low Speed

11. Hovering total-power-required data were acquired by stabilizing at 15^{o} heading intervals from 0^{o} to 360^{o} with respect to the wind. Power required near sea level in a 10-knot wind is shown in Figure D-1. IGE data for both maximum and minimum gross weights indicate

that more power was required during right crosswinds. The difference between maximum and minimum IGE power required was 50 horsepower (37.3 kilowatts) at 4150 pounds and 33 horsepower (24.6 kilowatts) at 3000 pounds. Maximum power required occurred in winds from 100° and 160° measured clockwise from the nose; minimum power required occurred in winds from 300° to 360° measured clockwise from the nose. Just the opposite wind effect is shown for OGE hover; that is, the data indicate that less power was required during right crosswinds. Maximum power required OGE occurred in winds from 240° to 300°, again clockwise from the nose; minimum power required occurred in winds from 100° and 120°. The difference between maximum and minimum OGE power required was 45 horsepower (33.6 kilowatts) at 4150 pounds and 28 horsepower (20.9 kilowatts) at 3000 pounds. The reason for the power-required differences between OGE and IGE due to wind effects is unknown; however, it is suspected that the fan and fin were less immersed in the main rotor downwash when hovering OGE, which allowed the wind to more strongly influence fan performance and fin weathercock characteristics. It is not clear why, when hovering IGE with the fan and fin immersed in the main rotor downwash, more power was required in right crosswinds and less power was required in left crosswinds, when just the opposite occurs OGE.

12. Forward and rearward flight power required is presented in Figure D-2. The data indicate that slightly more power was required in hover and in forward flight for OGE than for IGE, as expected; however at 40 knots rearward, IGE flight required 80 horsepower (14 percent of total usable) more than OGE flight. The data also indicate that more power was required for rearward flight than for forward flight for both IGE and OGE beyond 30 knots, and the disparity became greater as speed increased. A logical explanation for this is that the cambered fin was providing a force in forward flight which augmented fan thrust, and in rearward flight this force was either reduced or opposing fan thrust; hence, there was more engine torque.

13. Sideward-flight power required is presented in Figure D-3. These data indicate that more power was required in hover and right sideward flight for OGE than for IGE. Approximately the same power was required for IGE and OGE left sideward flight. More power was required for both IGE and OGE left sideward flight than for right sideward flight. It is possible that the fan must overcome weathercock stability in addition to main rotor torque during left sideward flight, whereas weathercock force augments fan thrust during right sideward flight. A similar effect was noted in tests of the Sikorsky fan-in-fin helicopter,[4] but for right sideward flight, since main rotor rotation was opposite to that of the SA-342.

14. Power required for 45° quartering flight is shown in Figure D-4. As in sideward flight, more power was required for flight to the left than to the right. It is also shown that more power was required for left quartering tailwinds than for left quartering headwinds, while the reverse was true, though not as significantly, for right quartering winds.

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[4] W. H. Meier, W. P. Groth, D. R. Clark, and D. Verzella, *Flight Testing of Fan-in-Fin Antitorque and Directional Control System and a Collective Force Augmentation System (CFAS)*, Sikorsky Aircraft Division, USAAMRDL-TR-75-19, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, June 1975.

Forward Flight

15. The power required for level flight, shown in Figure D-5, was recorded at a density altitude of 1850 feet and a gross weight averaging 3500 pounds (1587 kilograms). The minimum power required occurred at 62 KIAS. The maximum level-flight airspeed was 148 KIAS with the collective lever against the 15° elastic stop. For this condition, engine power was 520 horsepower (390 kilowatts), which was 50 horsepower (37.3 kilowatts) below the transmission limit. At the conditions tested, the helicopter was descending at 150 feet per minute at the 150-KIAS trim point.

16. The power required as a function of vertical speed is shown in Figure D-6 for 60, 90, 120, and 148 KIAS. The transmission limit of 570 horsepower (426 kilowatts) was reached at 120 KIAS and 1000 feet per minute climb with a collective pitch of 15° as measured at the root end of the blade. Part of the 50-horsepower (37.3 kilowatts) increase in total power required for the 120 KIAS climb condition over the 148 KIAS level-flight condition was apparently absorbed by the fan. The exact amount, however, was indeterminant. In the ball-centered climb, the pedals were positioned at 69 percent of total displacement (0 percent was full left and 100 percent was full right pedal); while at 148 KIAS level flight, the pedals were positioned at 43 percent of total displacement. Aerospatiale's flight test data, as shown in Figure A-5, indicates that the fan will absorb approximately 31 horsepower (23.1 kilowatts) more at a pedal position of 69 percent than at one of 43 percent.

Fan Power Required

17. The test day wind conditions were not suitable for determining accurate data on the power required to drive the fan. In order to minimize wind effects, data were acquired during stabilized hover at 15° intervals through a heading change of 360°, and were averaged. Table 3 shows these data for IGE and OGE hover and the percentage of fan power to total power required.

TABLE 3. FAN POWER REQUIRED FOR HOVER

Test Condition	Average Gross Weight (lb)	Density Alt (ft)	Average Total Power Required (hp)	Average Fan Power Required (hp)	% of Fan Power to Total Power Required
IGE	4150	-350	410	72.4*	17.6
OGE	4150	-250	415	69.7*	16.8
IGE	3000	-475	307	50.9*	16.6
OGE	3000	-375	311	50.3*	16.2
IGE	4100	7900	397	67.2	16.9
IGE	3950	9050	418	74.4	17.8

*These data were derived from pedal position and flight test data (Figure A-5).

These data indicate that more total power was required for hovering OGE than IGE, as expected; however, less fan power was required hovering OGE than hovering IGE. It is possible that when hovering IGE, the additional power required is due to the turning of the main rotor downwash through the fan duct (momentum drag), whereas the fan is less immersed in the downwash when hovering OGE. This power-required variation is in consonance with the findings reported in reference 1. The test data show that the fan power required (including fan gearbox losses) averaged approximately 17 percent of the total power required for hover. At 140 KIAS, the fan power was less than 5 horsepower (3.7 kilowatts). It is interesting to note that the total hover power required to weight ratio for all gross weights tested was approximately 1 horsepower per 10 pounds (1.0 kilowatt per 6.06 kilograms).

HANDLING QUALITIES

General

18. Stability and control tests were performed at the flight conditions listed in Table 2. The Model SA-342 exhibited slightly improved static lateral-directional stability over the Model SA-341 as reported in reference 1; however, the aircraft was still difficult to trim within the sideslip deadband. The trimming difficulty was attributed to a combination of high directional sensitivity, weak directional stability, and low fuselage sideloads. The improved fan provided sufficient control moment to attain 50 knots sideward flight. The response to directional control inputs in hover exceeded the requirements of MIL-H-8501A by a factor of 7. Following step inputs in a hover, the aircraft was slow in attaining a steady-state yaw rate. The aircraft was very stable and easy to control when hovering in left crosswinds or translating to the left, but translations to the right and precise hovering in right crosswinds were difficult to perform. This difficulty was also more pronounced when hovering IGE than when hovering OGE. With the improved fan, directional control effectiveness was very good in autorotative maneuvers. The aircraft's handling qualities were enhanced by extremely good speed stability and excellent turn coordination in cruise. The ride qualities throughout the test envelope were solid and comfortable, with the only significant roughness being a three-per-rev vibration that became noticeable at about 135 KIAS and increased with airspeed. There were no vibrations of any kind related to the fan, and the fan's overall operation was very smooth.

Forward Flight Characteristics

Trim Changes With Airspeed

19. The control positions required for trimmed, ball-centered, forward flight from 40 to 150 KIAS are shown in Figure D-7. The data were acquired at 1850 feet density altitude utilizing a sensitive airspeed measuring system. The pedal position varied from 61 percent of total control displacement at 40 KIAS to 44 percent of total displacement at 150 KIAS. From 40 to 80 KIAS, a gradual, nearly linear application of left pedal of 22 percent of the total displacement (0.86 inch) was required. From 80 to 130 KIAS, the pedal position remained essentially fixed at approximately 40 percent of total displacement. As the speed increased from 130 to 150 KIAS, a right pedal input of 5 percent of total displacement was required and level flight above 148 KIAS was no longer possible, as discussed in paragraph 15. The variation of longitudinal cyclic stick position with airspeed from 50 to 150 KIAS was stable and nearly linear. The stick moved forward from 43 percent to approximately 75 percent of the total displacement (0 percent was full aft and 100 percent was full forward cyclic) between 50 and 150 KIAS. Throughout the evaluation it was noted that the SA-342 possessed extremely good speed stability. This was exemplified by the ease of trimming on specific airspeeds and the minimal attention required for speed control. The data show minimal lateral stick migration with airspeed, remaining nearly constant at

41 percent of total displacement (0 percent was full left and 100 percent was full right cyclic), and the collective pitch followed the normal power-required curve.

20. The sideslip angle necessary for ball-centered level flight is shown in Figure D-8. As the airspeed increased from 40 to 100 KIAS, the sideslip angle decreased from -12^{o} to approximately 1.6^{o} and remained essentially constant thereafter. Aerospatiale stated that the best sideslip angle for cruise was about +3^{o} (right), which would cause the slip indicator (ball) to be displaced to the right by about one-half of its diameter. This was verified by the test results plotted in Figure D-9, which shows that the minimum power required occurred at a sideslip of +3^{o} at 120 KIAS. The same trend is shown for climbs and descents.

Trim Change With Vertical Speed

21. The trim change as a function of vertical speed (used as an indirect measure of power) was briefly examined in forward flight by varying collective pitch while maintaining ball-centered flight at 60, 90, and 120 KIAS. At a gross weight averaging 3500 pounds (1587 kilograms), the power was varied to produce vertical speeds of 0, ±500, and ±1000 feet per minute. The variation of control positions with vertical speed is shown in Figure D-10. The data show that the directional trim change between the descents at 1000 feet per minute and the climbs at 1000 feet per minute varied as much as 39 percent of total control displacement for the three speeds tested. Lateral cyclic stick position indicates minimal stick migration with vertical speed and was nearly constant at approximately 41 percent of total displacement. Longitudinal cyclic position again indicates a positive gradient with airspeed. A slight pitch-to-collective coupling was evidenced by a longitudinal trim change of as much as 20 percent of total control displacement between 1000 feet per minute descents and 1000 feet per minute climbs. Collective control position followed the typical power-required curve. The data show that collective pitch varied 4.5^{o} between the 1000 feet per minute descent and the 1000 feet per minute climb at 60 KIAS and 5.2^{o} between the two at 120 KIAS.

22. Figure D-11 shows the directional trim change discussed in paragraph 21 in terms of pedal position and engine power. Large pedal displacements were required to compensate for power. For example, at 90 KIAS, as the power was reduced from 450 horsepower (335.6 kilowatts) to zero (1000 feet per minute climb to autorotation), the pedals moved about 40 percent of the total control displacement. While the data were not complete, it would appear that at 90 KIAS, about 60 percent of the total pedal displacement would be required to compensate for a full power sweep (i.e., full power climb to autorotation). Although control margins were not a problem, this constituted more trim change than desirable and reflects the need for mechanical coupling of pedal with collective pitch, if it could be accomplished without adversely affecting the flat gradient of pedal position versus airspeed discussed in paragraph 19.

Lateral-Directional Stability

23. Steady heading sideslips were performed at 60, 90, and 120 KIAS with power as required for 0, ±500, and ±1000 feet per minute vertical speed. These tests were performed in this manner to study the effects of vertical speed on the lateral-directional stability characteristics. The variations of pedal position, lateral cyclic, and roll attitude with sideslip are shown in Figures D-12 through D-22. Figures D-23 and D-24 present the same information for 65 and 90 KIAS autorotation. Figures D-12 through D-15 show an abrupt

discontinuity in the pedal position versus sideslip data which occurred between the trim condition and the left sideslips. These discontinuities should have manifested themselves in an apparent loss of control effectiveness; however, no such change in control effectiveness was noted. Further, it appeared that the sideslip vane was sticking during the 60 KIAS tests near trim. Since the data from the 90 and 120 KIAS tests did not reflect this discontinuity, it is assumed that the sideslip vane was indeed sticking in the 60 KIAS dynamic pressure environment. In view of this assumption, Figures D-12 through D-15 are shown with a dashed line to reflect the probable gradient. The SA-342 exhibited slightly improved static directional stability over the SA-341 when compared to the information provided in reference 1; however, portions of the static stability gradients were still neutral or nearly so, as shown in Figures D-12 through D-24, and the attendant trimming difficulties still existed. This neutral region, which is referred to herein as a "sideslip deadband", varied in width from approximately $\pm 2^{\circ}$ to $\pm 8^{\circ}$ of sideslip about trim, depending upon airspeed and power. In addition to the weak directional stability within the deadband, the apparent dihedral effect, as reflected by roll attitude and lateral cyclic stick position, was neutral or very slightly positive. The data confirm the pilot comment of very sensitive directional control, especially when attempting to trim near zero sideslip, particularly in autorotation. Reducing power tended to shift the deadband to the right and to reduce the level of stability, while increasing airspeed narrowed the width of the deadband and increased the level of stability. The trimming difficulty was attributed to the combination of high directional control sensitivity, weak directional stability, and low fuselage sideloads. At sideslip angles beyond the deadband region, the static lateral-directional stability gradients became much more positive and trimming was much easier. It should be noted that the Sikorsky S-67 fan-in-fin helicopter exhibited similar characteristics.[4] Using the handling qualities rating scale shown in Appendix C, a rating of 5 is assigned the static lateral-directional stability characteristics.

Dynamic Stability

24. The directional dynamic stability of the SA-342 was evaluated by applying pedal pulses in trimmed cruise flight at 70, 100, and 150 KIAS. The stability was positive at all three airspeeds, although it appeared to deteriorate with increasing airspeed. At 70 KIAS, the response to the pulses was deadbeat; at 100 KIAS, an overshoot of 1/2 to 1 cycle was noted; and at 150 KIAS, the aircraft yawed about 2 cycles before the heading stabilized. Qualitatively, the weakening stability observed in the pulse testing was not significant, probably because of the short period of the oscillation (1.7 seconds at 150 KIAS) and the relatively small yaw angles (HQRS 2). For instance, a pedal pulse input of 18 percent (approximately 3/4 inch) produced an initial sideslip excursion from trim of only 7°, followed by peak-to-peak oscillations of about 8°, 6°, 4°, and 2°. Even though the sideslip excursions were small, relatively large fluctuations in engine torque accompanied these pedal-fixed yawing oscillations. For the sideslip excursions just described, the engine torque varied as much as 11 percent, which equates to approximately 60 horsepower (44.7 kilowatts). This variation of engine torque as a function of sideslip oscillation was also evident during the dutch roll discussed in paragraph 26.

25. The helicopter occasionally exhibited a slight unsteadiness in roll when in trimmed level flight, during gradual climbs and descents (up to 500 feet per minute), and at speeds above 90 KIAS. The unsteadiness was characterized by random, limit-cycle type roll oscillations of $\pm 1^{\circ}$ to $\pm 2\text{-}1/2^{\circ}$ with a period of about 2 seconds and was most prevalent in light turbulence. The same phenomenon was reported during previous flight testing of the SA-341. The aircraft always returned to its original trimmed roll attitude without corrective

inputs from the pilot. However, the data showed that when the pilot was in the loop, he was actively applying corrective lateral cyclic control. This unsteadiness was somewhat annoying, and its elimination would enhance the aircraft's handling qualities (HQRS 3).

26. An occasional well-damped lateral-directional oscillation resembling a dutch roll was observed in trimmed cruising flight in light turbulence; and a well-defined neutral to lightly damped dutch roll was encountered in right, steady-heading sideslips at 90 and 120 KIAS. At 120 KIAS and +14° of sideslip, the dutch roll was characterized by a clockwise motion of the nose with pitch and yaw excursions of 2° to 3°, a roll-to-yaw ratio of about 1, and a period of 1.9 seconds. The motion noted in cruise was considered to be insignificant, and the dutch roll in the sideslips occurred only at sideslip angles greater than 10°, so it, too, was of little significance since its presence was well outside the normal operating regime.

27. The SA-342 exhibited excellent turn coordination characteristics in cruising flight. It was possible to make well-coordinated cyclic-only turns with the feet off the pedals, provided the aircraft was properly trimmed initially. The helicopter did not have a pedal centering feature, but the pedals remained fixed wherever positioned. Cyclic only rapid rolling reversals to ±45° bank angle at 120 KIAS revealed an adverse yaw or sideslip lag of only about 5° (HQRS 1.5).

Maneuvering Stability

28. The maneuvering stability characteristics were qualitatively assessed between normal load factors of approximately 0.2 and 2.0 at airspeeds up to 150 KIAS. The variation of longitudinal cyclic stick position with load factor was positive, with no detectable flattening of the gradient or "dig in" tendency. Also, within the operating envelope of the hydraulic control system, there was no noticeable lateral cyclic stick migration or tendency for the aircraft to roll as a function of load factor. Lateral control was assessed during pushovers and was found to be adequate. The operating envelope of the hydraulic control system was defined by those flight conditions where main rotor control loads (feedback forces) did not exceed the force capability of the actuators. Exceeding the envelope saturated the system and caused feedback forces in the cyclic stick which tended to roll the aircraft smartly to the right. This was experienced in the test helicopter at about 2g's and 140 KIAS. The transient involved in exceeding the limit was undesirable, particularly for a nap-of-the-earth environment. Aerospatiale indicated that the production SA-342 will have a higher capacity hydraulic control system. The maneuvering stability characteristics of the SA-342 were assigned an HQRS of 2 within the operating envelope of the hydraulic control system; however, the transient associated with exceeding the envelope degraded the rating to a 6.

Autorotations

29. Autorotation entries, maneuvering descents, and touchdowns were performed to evaluate the directional control effectiveness provided by the improved fan. Inadequate directional control for performing coordinated autorotative maneuvers had been previously reported for the SA-341 in reference 1. Yaw excursions during autorotative entries for the SA-342 were mild, although close attention was needed to keep the ball centered as the aircraft transitioned from powered to autorotative flight. Good rotor speed and airspeed stability were noted during the descents. Control effectiveness was very good throughout the autorotative maneuvers; but as discussed previously and as shown in Figures D-23 and D-24, the aircraft was very sensitive directionally, making it easy to overcontrol when

attempting to trim for a given sideslip angle. The high yaw sensitivity and the opposite-turning rotor (compared to U. S. helicopters) did not present any problems in controlling the aircraft during autorotative landings. In rolling and turning the aircraft to bank angles as high as 70° during 65 KIAS autorotative descents, the pedal position varied from 25 to 37 percent of total control displacement. The pedal position for trim was approximately 30 percent. This illustrated a marked improvement over the SA-341 as discussed in reference 1. Normal type autorotative landings into a 5- to 8-knot headwind required pedal inputs which increased from 32 to 50 percent of total control displacement during the flare, and then decreased from 50 to 10 percent of total control displacement as collective pitch was applied to arrest the sink rate and to complete the landings. The landings were made near sea level standard conditions at a gross weight of approximately 3000 pounds (1365 kilograms). The improved fan provided adequate control for maneuvering the aircraft during autorotative descents, and the control margin in landing appeared to be adequate, although the adverse effects of high density altitude and right crosswinds were not evaluated. Also, the effect of the cambered fin on autorotation entries at high speed was not examined.

Pitch and Roll Due to Yaw

30. A brief investigation of the pitch and roll due to yaw characteristics was made at 60 KIAS by applying right and left directional control inputs while holding the cyclic stick fixed. The inputs varied from 7 to 18 percent of total control displacement, which was sufficient to yaw the aircraft beyond the sideslip deadband region discussed in paragraph 23. The SA-342 exhibited positive apparent dihedral effect in both directions, being significantly stronger to the left. The aircraft, in response to the pedal step inputs, began an immediate roll in the direction of the input, which assumed a steady rate within about 2 seconds. Extrapolated data for 1-inch inputs (approximately 25 percent of total control displacement) indicated roll rates of about 10° per second to the right and 15° per second to the left. The aircraft also exhibited a pitch due to yaw that was almost of the same magnitude as the roll due to yaw. A pitch-up was associated with right pedal step inputs, and a pitch-down was associated with left step inputs.

Boost-Off Characteristics

31. The flight control hydraulic system was secured (simulated boost failure) during 130 KIAS cruising flight, and a boost-off approach to hover was performed. The control forces required to come to a hover were estimated to be less than those defined in paragraph 3.5.8(a)(2) of MIL-H-8501A; however, the forces could not be trimmed out, since no trim capability was provided. The transient associated with the simulated loss of hydraulic pressure was abrupt and potentially dangerous. The simulated loss of the power-operated system (or saturation as discussed in paragraph 28) allowed control loads to feed back into the cyclic stick. These forces tended to move it aft and to the right. The resultant stick motion, if not restrained, caused a rapid right roll and a moderate pitch-up and right yaw of the helicopter. The roll transient far exceeded the limits defined in paragraph 3.5.8(a)(1) of MIL-H-8501A. Even with the pilot properly grasping the stick, but not necessarily anticipating the event, the loss of hydraulics caused a significant attitude change. This characteristic was unsatisfactory and received an HQRS of 6. Aerospatiale stated that the production SA-342 will have a hydraulic accumulator and a visual warning system, which should allow the pilot time to anticipate the transient effects of a hydraulic failure and prevent hazardous attitude changes.

HOVERING AND LOW-SPEED CHARACTERISTICS

Angular Response

32. Directional control step inputs were applied while hovering both IGE and OGE. These tests were performed in calm winds at a gross weight averaging 3700 pounds (1680 kilograms) and with a mid center of gravity. The step inputs varied in size from 8 to 16 percent of total control displacement (about 1/4 to 2/3 inch) and were held constant until the aircraft yawed 360° or until the yaw rate became excessive. The only directional response parameter recorded was aircraft gyro-stabilized heading. Although limited angular rate and acceleration data were derived from the heading trace, the data necessary to properly define transport lag were not available.

33. The SA 342 was extremely responsive in yaw. The heading change after the first second, the steady-state yaw rate, and the maximum angular acceleration are shown as a function of control input in Figure D-25. Initial responses to the pedal steps were exceptionally crisp, with heading changes in the first second ranging from 14° to 31°. Figure D-25 shows that if a straight line were extended through the data points, a 1 inch right pedal step would produce a heading change of 48° in the first second. This would approach the maximum "control power" of 50° in the first second specified in paragraph 3.3.7 of MIL H 8501A (for the lightest normal service loading), and it would exceed the 6.6° minimum specified in paragraph 3.3.5 of MIL-H-8501A (for the maximum overload gross weight) by a factor of 7. Although not tested, it is likely that the aircraft would have yawed somewhat faster at the lightest normal service loading and slightly slower at the maximum weight. The steady-state yaw rates generated by the step inputs ranged from 0.8 to 2.1 radians per second. A yaw rate of 1.5 radians per second and an acceleration of 0.6 radian per second squared were generated by a right pedal step of 1/2 inch (12.7 percent of total displacement). A 1/2 inch left pedal step produced a rate of approximately 1.0 radian per second with an angular acceleration of 0.4 radian per second squared. Extrapolated data from Figure D 25 would indicate a control power of 1.1 radians per second squared to the right and directional control sensitivities of approximately 1.5 and 1.1 radians per second squared per inch, right and left, respectively. The high apparent directional sensitivity of the SA 342 was clearly related to its control system gain. The total pedal displacement available was only 3.9 inches as compared to 6.5 inches for the UH-1, 6.8 inches for the OH-58, and 7.6 inches for the OH-6 helicopters. If the effect of changing the gain were proportional, increasing the SA-342 total pedal displacement available to 7 inches would bring its responses to 1-inch steps to well within the maximum and minimum values identified in paragraphs 3.3.5 and 3.3.7 of MIL-H-8501A. Also, the response to a full right pedal step input with the desensitized gain would be very close to that specified in paragraph 2.3.5 of MIL-H-8501A (neglecting the weight delta). This calculated response resulting from an increase in total pedal displacement available is shown in Figure D-25 as a dashed line. This change might improve the trimming difficulties encountered within the sideslip deadband in cruising flight discussed in paragraph 23, and it might also lessen the pilot effort required for hovering to the right and in right crosswinds discussed in paragraphs 35 and 36. Little difference was noted between the right and left initial directional responses (heading change during the first second). However, the data show that for equal pedal inputs of 10 percent of total control displacement, the aircraft developed a 50 percent greater steady-state yaw rate to the right than to the left. The data presented in Figure D-25 indicate a possible nonlinearity or degradation of steady-state yaw rate per unit of control input at the larger values of control input.

34. The helicopter was very slow in developing steady-state yaw rates following the step inputs. This lag appeared to the pilot as an undesirable directional windup. The Sikorsky S-67 helicopter with the fan-in-fin system exhibited a similar characteristic. Four typical yaw rate curves constructed from the heading traces are shown in Figure D-26. Based upon the limited data acquired, the time required to achieve a steady state yaw rate averaged about 6 seconds for the SA-342, whereas the S-67 time lag averaged about 8.5 seconds. The SA-342 yaw rate curves shown in Figure D-26 illustrate the nature of the apparent weak directional damping and indicate that there was slightly less apparent damping to the right than to the left. (This apparent weak damping may have been the result of the aircraft's directional damping characteristics, variations in fan efficiency, or a combination of both.) These data show an average time constant (τ) of 2.7 seconds, which corresponds to a directional damping value (τ^{-1}) of 0.37 per second. The directional response characteristics, as related to the windup effect, can best be summarized from an operational standpoint, in terms of the time required for the aircraft to assume a new heading or the time to yaw through a given angle from a steady trimmed hover, as shown in Figure D-27 as a function of control input. The figure shows that the aircraft performed a 180° hovering turn to the right in 3 seconds with a pedal displacement of 0.6 inch (15.2 percent of total control displacement). Figure D-26 shows that during the turn, the yaw rate was rapidly building to a rate of over 90° per second. To counter the windup effect, the pilot should reduce the pedal input as the turn progresses. However, should these high rates be allowed to occur, they must be arrested carefully, especially when stopping left turns wherein transmission or engine overtorques can occur. While the windup characteristic was not bothersome during mild maneuvering, it could become undesirable for those missions which require high agility. The poor directional damping and high sensitivity characteristics degraded the handling qualities (HQRS 5).

Hovering Translations

35. Cyclic, collective, and directional control positions for the hovering translations discussed in paragraphs 12, 13, and 14 are shown in Figures D-28, D-29, and D-30. Figure D-28 shows that more right pedal was required at 40 knots rearward than for 40 knots forward, which further supports the discussion in paragraph 12. The improved fan provided sufficient control moment to attain 50 knots in sideward flight at a gross weight of 3700 pounds (1680 kilograms). For OGE sideward flight, full right pedal was required for 50 knots to the left, while a 7 percent left pedal margin remained at 50 knots to the right. For IGE sideward flight, a 2 percent right pedal margin existed at 50 knots to the left and a 15 percent left pedal margin remained at 50 knots to the right. Translations to 40 knots in the 45° directions were performed with more than adequate control margins. Pilot effort varied considerably depending upon the direction and velocity of flight. Figure D-31 is a polar diagram that represents level of pilot effort as a function of translational speed and direction. The helicopter was extremely stable and easy to fly in the unshaded region of the diagram. Translations in the shaded regions required continuous corrective cyclic, collective, and directional control inputs. The parabolic-shaped area represents the region where the usual control difficulty is encountered with single-rotor helicopters in rearward flight or hovering with a tailwind. The data show that control activity diminished slightly as translation speeds to the right approached 40 knots, although the pilot was not aware of such a trend. In contrast to an HQRS of 2 that describes the handling qualities associated with hovering translations to the left, an HQRS of 5 applies to the pilot effort required in translations to the right and to the rear.

Turns Over a Spot

36. Figure D-32 shows the pedal positions required for the stabilized hovering at various wind azimuths described in paragraph 11. The test results showed the critical wind azimuth to be about 290° as measured clockwise from the nose. An 8 percent right pedal margin was the least margin encountered during turns over a spot. This occurred while performing IGE hovering turns at a density altitude of 9050 feet, with a gross weight of 3950 pounds (1790 kilograms), and in a wind estimated at 5 to 10 knots. Complementary to the findings discussed in paragraph 35, the aircraft was very stable and easy to control in left crosswinds, but precise hovering with the wind coming from the right and rear quadrants was relatively difficult to perform. Wind coming from 60° to 90° of the nose (measured clockwise from the nose) caused the most problem, and hovering IGE was considerably more difficult than hovering OGE. These unstable regions were characterized by frequent, rather abrupt disturbances in all axes, with roll being the most stable. The pitch and yaw excursions appeared to occur somewhat in consonance, as if disturbed main rotor downwash might be impinging upon the horizontal tail and then spilling randomly through the fan. A significant amount of collective control activity was also required in the unstable region. In contrast to an HQRS of 2 for hovering in a left crosswind, an HQRS of 5 applies to the pilot effort required to hover in a right crosswind.

Takeoff and Landing Characteristics

37. Takeoff and landing characteristics were observed in making normal vertical lift-offs and landings, including autorotative touchdowns, over various terrain features including slopes. The skid type landing gear featured a single-point pivot-type support for the aft mounting, which in effect softened the gear. This design tuned the landing gear in conjunction with the elastomeric lead lag rotor blade dampers to avoid ground resonance. Within the scope of the evaluation, no tendency toward ground resonance was noted; but the soft, pivoted gear caused the helicopter to be very sensitive to pedal movements while sitting on the ground with rotor turning. Small pedal movements produced an unusual rolling motion in the fuselage; however, it did not bother the actual lift-offs or autorotative landings, and it had no effect on the slope landings as had been anticipated. The slope landing capability of the SA 342 was outstanding. A unique, floating, main rotor blade droop stop ring allowed landings to be made on slopes as steep as 15° with no increased vibration, droop stop pounding, or mast bumping. The only adverse aspect of the landing characteristics was the general aircraft unsteadiness in hover; consequently, considerable control activity was required to perform smooth vertical landings. This characteristic was not unlike that of the U. S. Army's OH 58 helicopter. The overall takeoff and landing characteristics of the SA 342 were assigned an HQRS of 3.5.

CONCLUSIONS

38. Within the scope of the limited evaluation of the SA-342 helicopter, the following conclusions were made:

a. The total power required for hovering turns in a 10-knot wind varied as much as 50 horsepower (37.3 kilowatts) depending upon the direction of the wind (paragraph 11).

b. Significant contrasting differences existed in the total power required for IGE and OGE hover between right and left crosswinds (paragraph 11).

c. Significantly more total power was required for rearward flight than for forward flight both IGE and OGE beyond 30 knots (paragraph 12).

d. More total power was required in left sideward flight than in right sideward flight (paragraphs 13 and 14).

e. The minimum power required in level flight occurred at 62 KIAS, and full collective pitch (15^{o}) produced a level-flight indicated airspeed of 148 KIAS (paragraph 15).

f. The fan power required in hover was approximately 17 percent of the total power required, with slightly less fan power required for OGE hover than for IGE hover. Fan power required in level forward flight at 140 KIAS was less than 5 horsepower (3.7 kilowatts) (paragraph 17).

g. The pedal position was nearly constant from 80 to 130 KIAS, and lateral cyclic stick migration with airspeed was negligible (paragraph 19).

h. The aircraft exhibited extremely good speed stability (paragraph 19).

i. Directional trim change as a function of power indicated the need for pedal to collective pitch coupling (paragraph 22).

j. The SA-342 exhibited slightly improved static lateral-directional stability over the SA-341 as reported in reference 1; however, the aircraft was still difficult to trim directionally within the sideslip deadband. The trimming difficulty was attributed to the combination of high directional sensitivity, weak directional stability, and low fuselage sideloads (paragraph 23).

k. Directional dynamic stability in forward flight was positive, but it appeared to deteriorate with increasing airspeed. Large fluctuations in engine torque accompanied pedal-fixed yawing oscillations (paragraph 24).

l. The SA-342 possessed excellent turn coordination characteristics in cruising flight (paragraph 27).

m. Maneuvering stability was positive, and lateral control during pushovers was adequate (paragraph 28).

n. The improved fan provided adequate directional control moment and margin during autorotative maneuvering (paragraph 29).

o. The cyclic stick force transients resulting from a loss of the hydraulic flight control system or from saturation of the system in maneuvers were undesirable and exceeded the limits defined in paragraphs 3.5.8(a)(1) and (2) of MIL-H-8501A (paragraph 31).

p. Aircraft response to directional control inputs in hover exceeded the minimum requirements of paragraph 3.3.5, MIL-H-8501A by a factor of 7. An increase in total pedal displacement would improve the directional sensitivity characteristics (paragraphs 32 and 33).

q. The aircraft possessed weak apparent directional damping in hover, which manifested itself as a directional windup (paragraph 34).

r. The improved fan provided sufficient control moment to attain 50 knots in sideward flight (paragraph 35).

s. The aircraft was very stable and easy to control when hovering in left crosswinds or translating to the left, but precise hovering in right crosswinds or translations to the right were difficult to perform. The unsteadiness was also more pronounced when hovering IGE than OGE (paragraphs 35 and 36).

t. The aircraft exhibited good slope landing capability (paragraph 37).

39. In summary, the handling qualities of the SA-342 were generally very good with the exception of a region of poor lateral-directional stability in forward flight, poor directional damping in hover, high directional sensitivity, excessive directional trim change as a function of power, and a pronounced pitch, yaw, and vertical unsteadiness in certain hovering regimes. The helicopter exhibited excellent speed stability and turn coordination characteristics in cruise flight. The improved fan provided more directional control power than normally found on helicopters with tail rotors, and its operation was very smooth with no vibration whatsoever. The maneuvering stability was positive and lateral control during pushovers was adequate. Cyclic stick force transients associated with saturation or failure of the hydraulic control system were excessive. The data indicated that a performance penalty was incurred in hover due to the fan as compared with typical tail rotors. The mechanical simplicity of the fan and the safety aspect of having the fan protected by the fin constitute strong trade-offs in favor of the fan-in-fin antitorque device.

RECOMMENDATIONS

40. It is recommended that additional effort be expended to explore and resolve the problems associated with the fan-in-fin antitorque/directional control systems which are identified in this report and which have been reported in references 1 and 4, in particular the poor lateral-directional stability in forward flight and the poor apparent directional damping in hover. Additional testing is also recommended to obtain precise fan thrust and corresponding fan power-required data in the flight environment.

REFERENCES

1. Kelley, H., and West, T. C., *Flight Investigation of Effects of a Fan-in-Fin Yaw Control Concept on Helicopter Flying Quality Characteristics*, NASA TN D-7452, Langley Directorate, U. S. Army Air Mobility Research and Development Laboratory, Hampton, Virginia, April 1974.

2. *Helicopter Stability and Control*, Flight Test Manual, Naval Air Test Center, USNTPS-FTM-No. 101, 10 June 1968.

3. *Helicopter Performance*, Flight Test Manual, Naval Air Test Center, USNTPS-FTM-No. 102, 28 June 1968.

4. Meier, W. H., Groth, W. P., Clark, D. R., and Verzella, D., *Flight Testing of Fan-in-Fin Antitorque and Directional Control System and a Collective Force Augmentation System (CFAS)*, Sikorsky Aircraft Division, USAAMRDL-TR-75-19, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, June 1975.

APPENDIX A
DESCRIPTION OF THE SA-342

GENERAL

The Model SA-342 helicopter was a derivative of the Model SA-341. Significant differences were noted in paragraph 4 of the Introduction of this report. A general description of the SA-342 is contained in the following paragraphs. A two-dimensional sketch is shown in Figure A-1. Physical characteristics are summarized in Table A-1.

MAIN ROTOR

The main rotor consisted of three fully articulated, manually foldable blades. The rotor hub and mast were a single unit. Each blade was attached to its respective feathering bearing housing by a lead-lag hinge and an elastomeric lead-lag damper. The feathering bearing housings were attached to the hub by flapping hinges. The blades were constructed of plastic and fiberglass and were reinforced with honeycomb filler. Rotor diameter was 34.45 feet (10.5 meters).

TAIL FAN

The fan, shown in Figure A-2, had 13 die-forged aluminum alloy blades with an asymmetric airfoil section (63A2XX, thickness distribution unknown). The diameter of the fan was 27.36 inches (0.695 meter), while the diameter of the shroud at the fan plane was 27.56 inches (0.700 meter). Rotational speed of the fan was 6000 rpm. Fan blade pitch range was $-24^{\circ}49'$ to 40°. The blades were attached to the hub by pitch bearings and were sealed as shown in Figure A-3 to prevent spanwise flow emanating from the centerbody. Figure A-4 shows the previous SA-341 blade and hub configuration. The diameter of the centerbody at the fan plane was 12.79 inches (0.325 meter). Fan power required as a function of pedal position is shown in Figure A-5.

FIN

The SA-341 and the SA-342 had the same fin design as shown in Figure A-6. The section of the fin above the fan was twisted and cambered with the intention of unloading the fan in forward flight. The airfoil sections of the upper fin were not standard. The manufacturer provided the following information concerning fin geometry: 4 percent camber, 18 percent thickness at the manufacturing break (located immediately above the fan shroud) varying linearly to a 12 percent thickness at the tip, and a 2° incidence angle. The effective fin area was approximately 21.5 square feet (2.0 square meters) including the duct. In addition, the vertical fins mounted at the ends of the horizontal stabilizer had a combined area of 5.4 square feet (0.5 square meter). The section of the fin below the fan was symmetrical. Aerospatiale tests indicated that at 0° angle of attack, the fin lift coefficient was approximately 0.2, the angle of zero fin lift was -3.5°, and the lift curve slope was about 0.053.

TABLE A-1. BASIC AIRCRAFT INFORMATION

GENERAL	
Takeoff gross weight for tests	3750 lb (1700 kg) to 4333 lb (1965 kg)
Overall length (not counting nose boom)	31.37 ft (9.5 m)
Overall height (ground to top of fin)	10.21 ft (3.1 m)
Landing gear tread	6.6 ft (2.0 m)
Engine power	858 hp (640 kw)
Seating capacity (including pilot and copilot)	5
MAIN ROTOR	
Diameter	34.45 ft (10.5 m)
Number of blades	3
Blade chord	11.81 in. (30 cm)
Airfoil section	NACA 0012
Twist	-6^{o}
Blade taper ratio	1
Disk area	934.12 ft^2 (86.6 m^2)
Blade area	50.89 ft^2 (4.73 m^2)
Solidity	0.055
Normal operating speed	387 rpm
Rotational speed limits	82 to 114 pct (317 to 441 rpm)
TAIL FAN	
Diameter	27.36 in. (0.695 m)
Number of blades	13
Blade chord	1.54 in. (3.9 m)
Airfoil section	NASA 63A2XX
Twist	-7^{o}
Pitch angle range	-24^{o} 49' to 40^{o}
Blade area	273.42 $in.^2$ (1764 cm^2)
Disk area	589.78 $in.^2$ (3805 cm^2)
Solidity	0.46
Normal operating speed	6000 rpm
FIN	
Total area including fins at ends of horizontal stabilizer (excluding duct area)	22.8 ft^2 (2.1 m^2)
Angle of attack for zero lift	-3.5^{o}
Estimated lift-curve slope	0.053
Lift coefficient at zero angle of attack	0.02
Incidence	2^{o} (measured ccw from nose)
Airfoil (at manufacturing break just above duct)	NACA 4418 (modified)
Airfoil (at upper tip)	NACA 4412 (modified)
DUCT	
Area of duct	4.14 ft^2 (.385 m^2)
Diameter at fan	27.56 in. (.700 m)
Diameter of centerbody	12.79 in. (.325 m)
Length	11.69 in. (.297 m)
TOTAL CONTROL DISPLACEMENT (at grip centers)	
Lateral cyclic	9.45 in. (.240 m)
Longitudinal cyclic	11.46 in. (.291 m)
Pedals	3.94 in. (.100 m)
Collective	16.8^{o} (normal range was 15^{o} with a 1.8^{o} contingency above elastic stop)

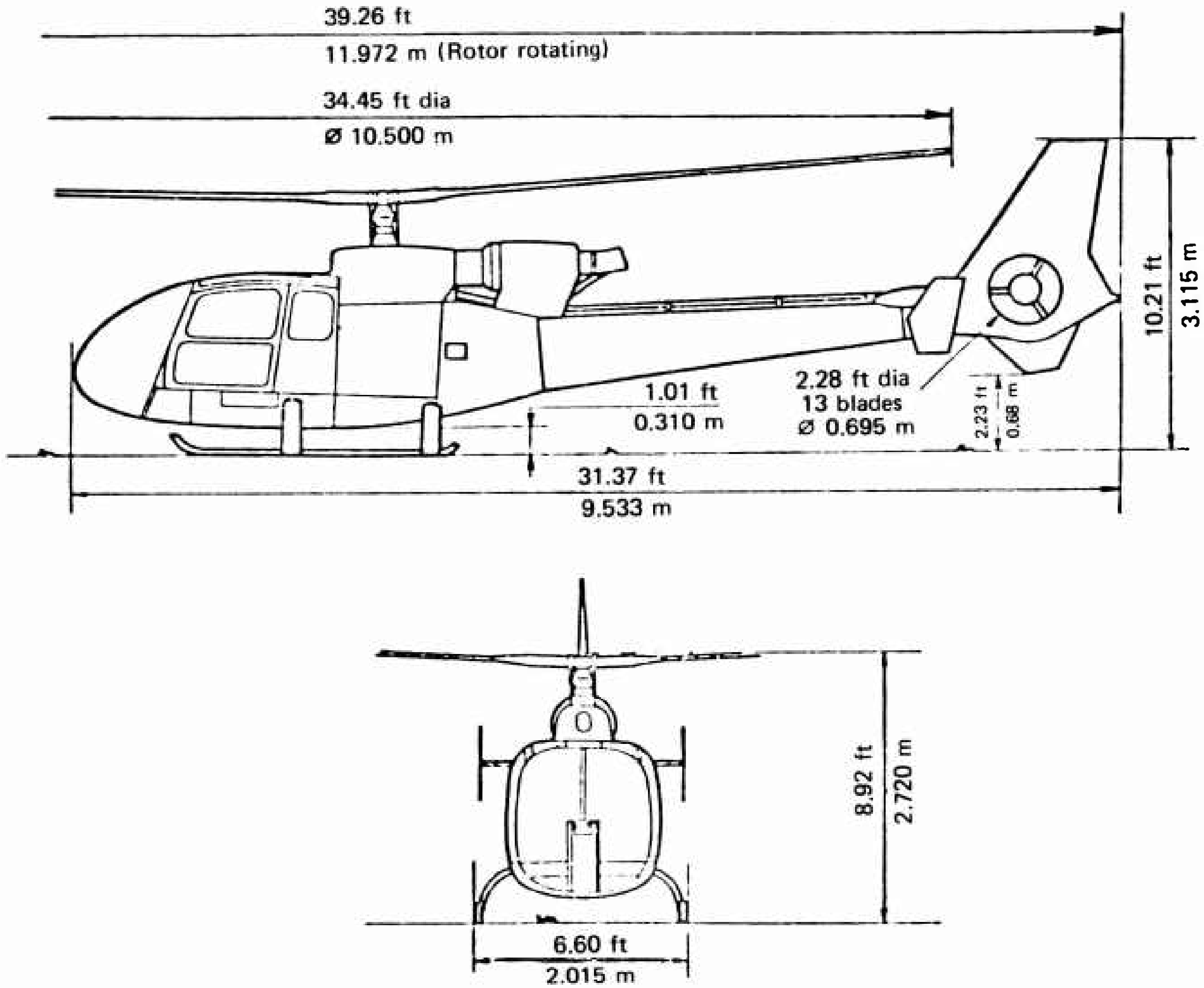


Figure A-1. Main dimensions of the SA-342 in flight configuration.

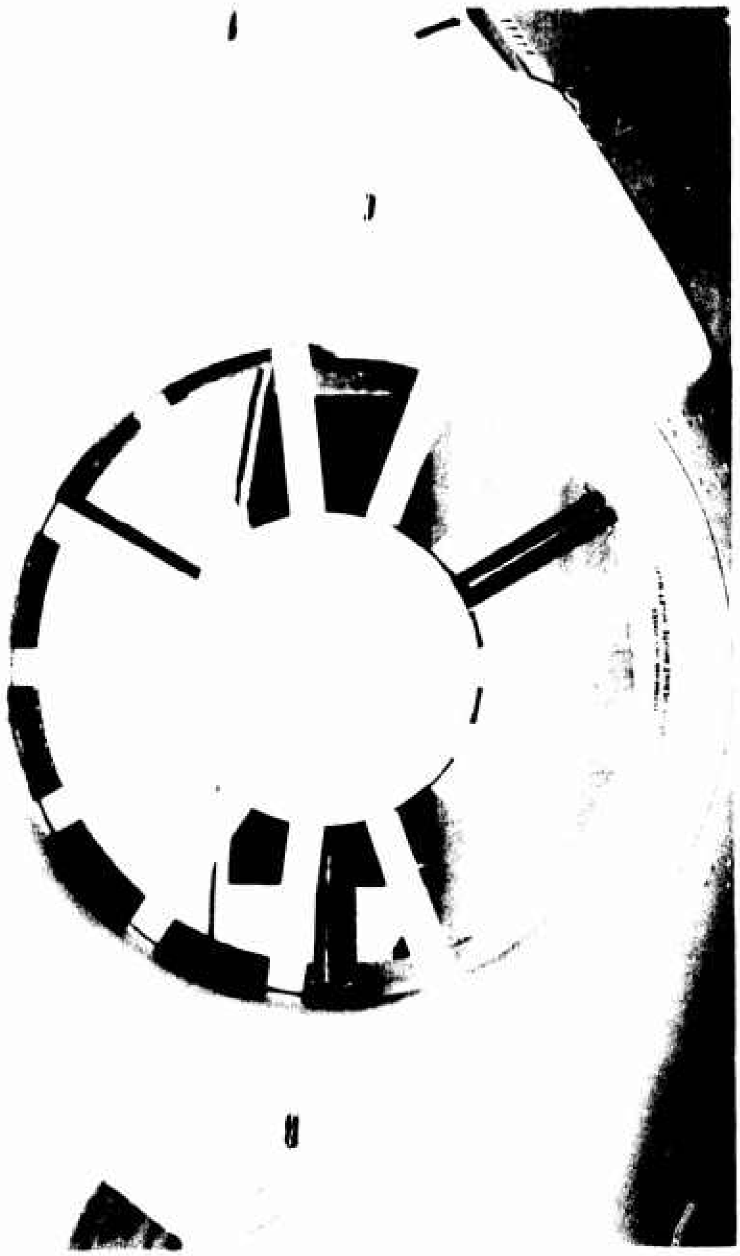


Figure A-2. Planform view of SA-342 improved fan, intake side.

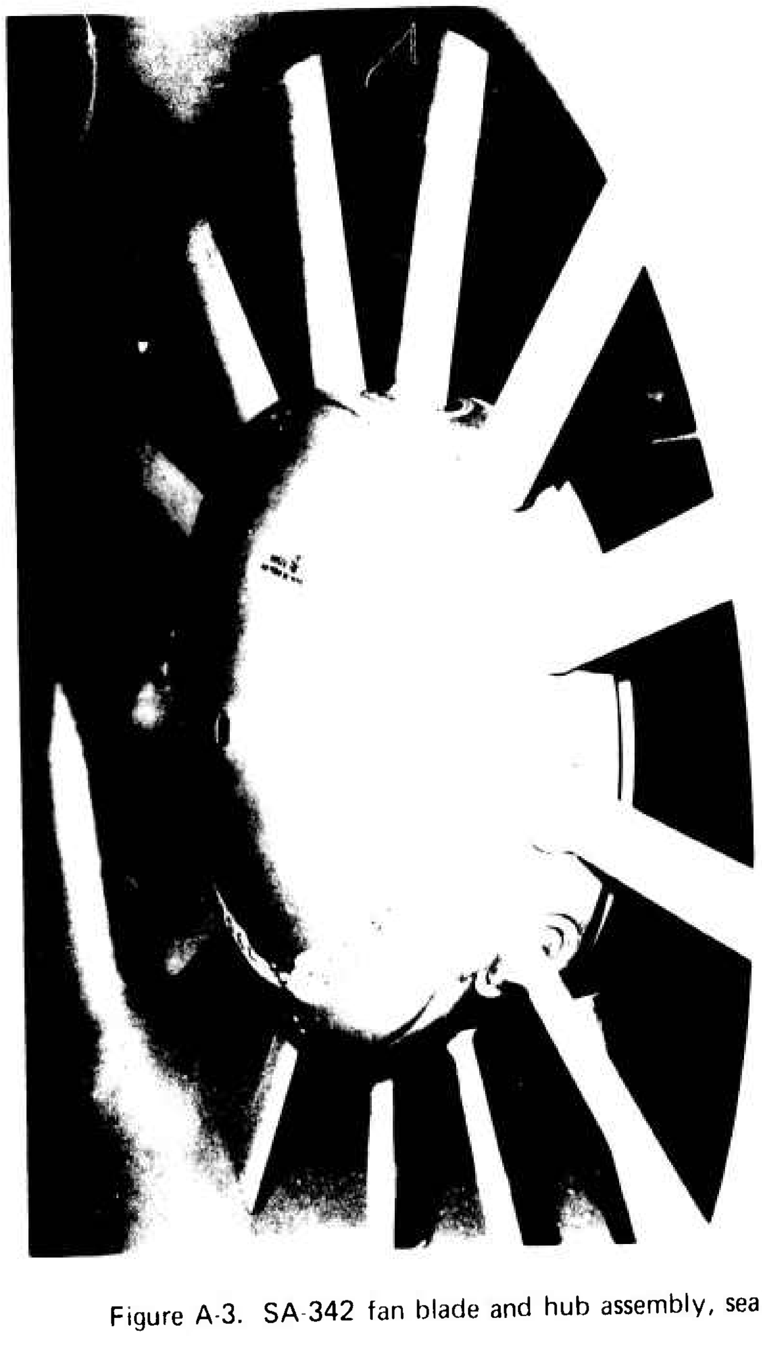


Figure A-3. SA-342 fan blade and hub assembly, sealed.

Figure A-4. SA-341 fan blade and hub assembly, unsealed.

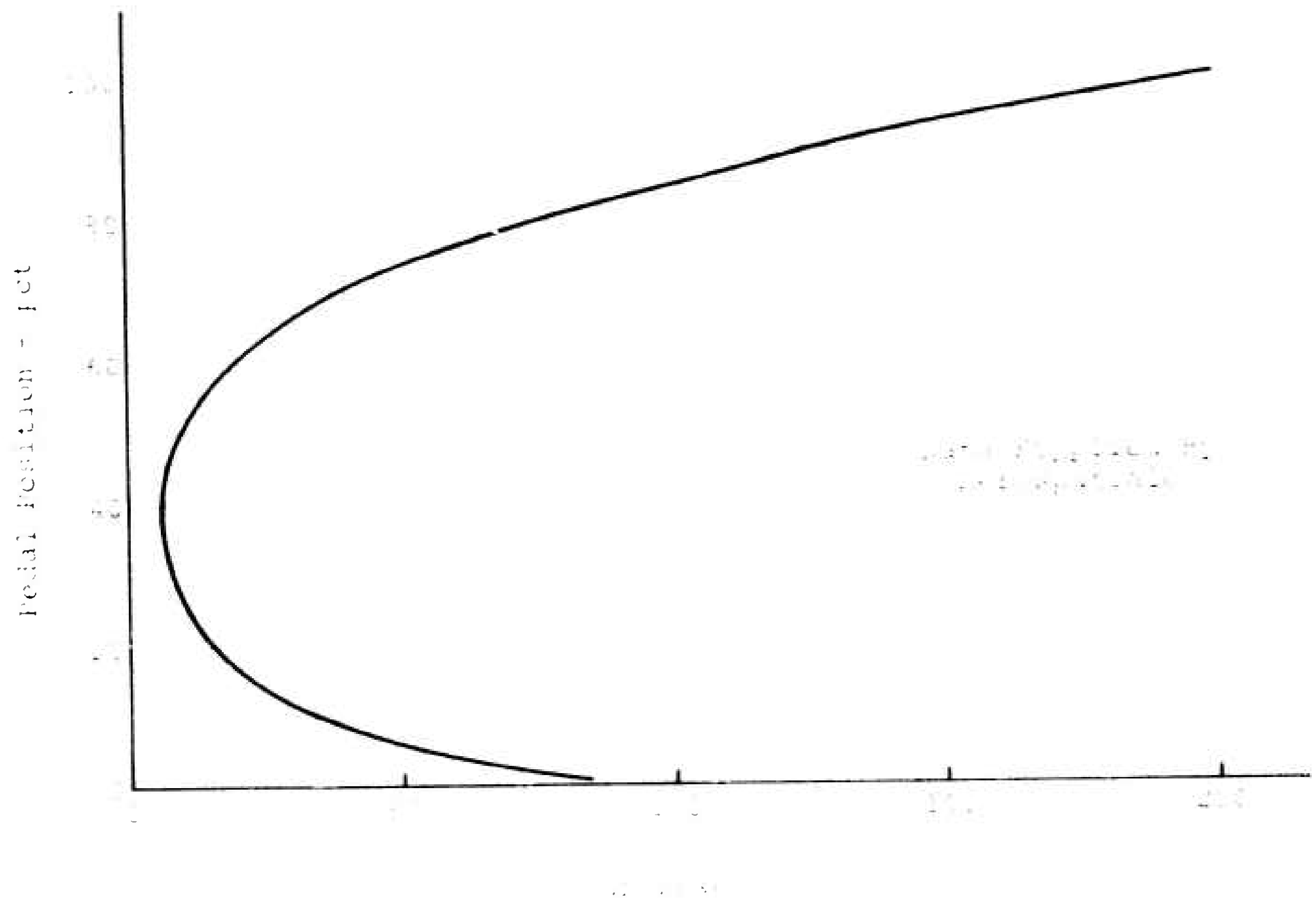


Figure A-5. Variation of fan power required with pedal position (flight test data).

Figure A-6. SA-341 and SA-342 empennage.

LANDING GEAR

The landing gear was of a steel tube skid design with provisions for ground handling wheels, floats, or skis. The gear featured a single-point pivot-type support for the aft mounting which in effect softened the gear. Aerospatiale stated that this feature was added to prevent ground resonance problems. Paragraph 37 of the Results and Discussion section of the report discusses this characteristic in more detail.

AIRFRAME

The cockpit structure was basically a welded light alloy frame, which housed the doors and windows, and was mounted on a lower structure of two longitudinal box sections connected by frames and bulkheads. The central section supported the transmission and housed the baggage hold and fuel tank. The rear section supported the engine and tailboom. The cockpit floors and the central and rear sections were constructed of aluminum honeycomb sandwich panels. The tailboom, horizontal stabilizer, and fin were of conventional sheet metal construction. The cockpit normally accommodates five persons, but weight and size of the instrumentation for the test vehicle reduced the seating capacity to three.

APPENDIX B
TEST INSTRUMENTATION

GENERAL

All instrumentation, exclusive of the voice recorder, was installed, calibrated, and maintained by the manufacturer at the test site.

VOICE RECORDED DATA

Cockpit quantitative data and pilot qualitative comments were recorded via a cassette recorder provided and operated by the Eustis Directorate, USAAMRDL test team. Parameters recorded in this manner included:

- Ground pressure altitude
- Barometric altitude (ship system)
- Outside air temperature
- Airspeed (ship system)
- Vertical speed
- Sideslip angle (boom system cockpit readout)
- Pitch attitude (ship system)
- Roll attitude (ship system)
- Heading (ship system)
- Engine torque
- Rotor speed
- Pedal position (instrumentation cockpit readout)
- Collective position (cockpit indicator)
- Fuel quantity

OSCILLOGRAPH DATA

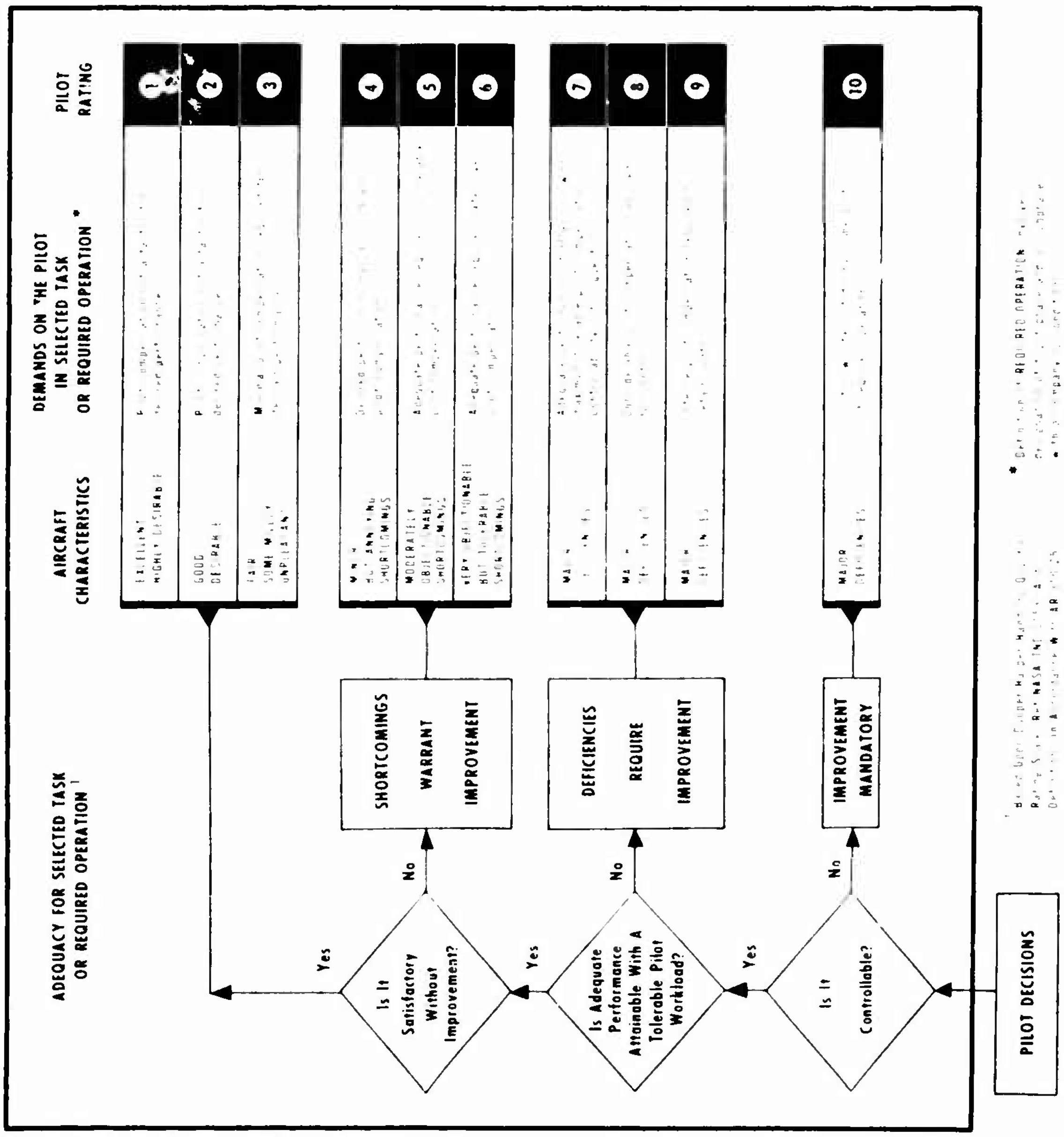
Quantitative data were recorded on the aircraft's oscillograph system. The following test parameters were recorded:

- Barometric altitude (boom system)
- Airspeed (boom system)
- Time
- Pitch attitude
- Roll attitude
- Heading
- Sideslip angle
- Longitudinal cyclic control position
- Lateral cyclic control position
- Directional control position
- Collective lever position

OSCILLOGRAPH RECORDED DATA (Continued)

Rotor speed
Engine torque
Tail fan servo displacement
Tail fan drive shaft torque (Flight No. 5 only)

APPENDIX C
HANDLING QUALITIES RATING SCALE



APPENDIX D
TEST DATA, SA-342

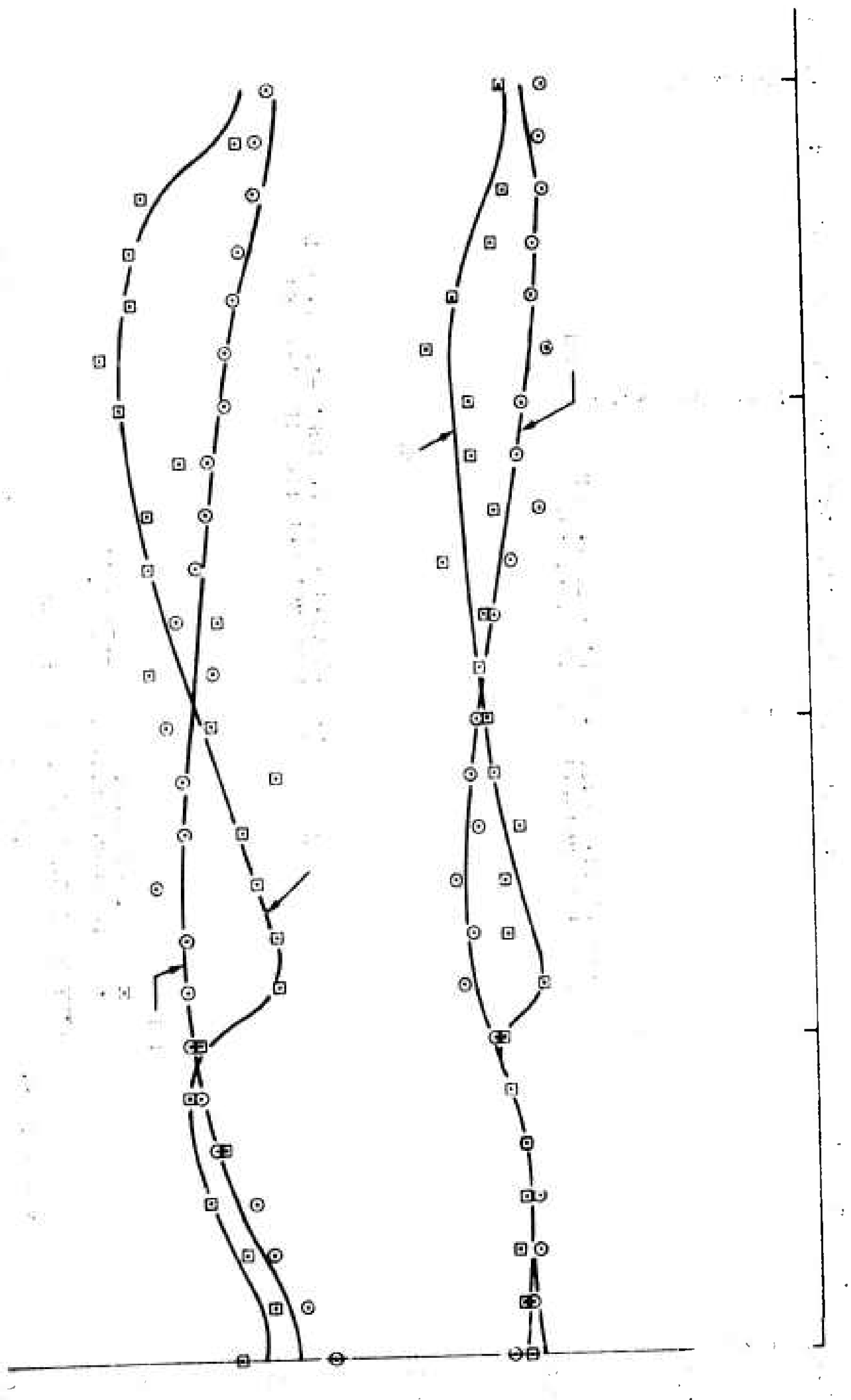


Figure D-1. Hovering power required at various wind azimuths.

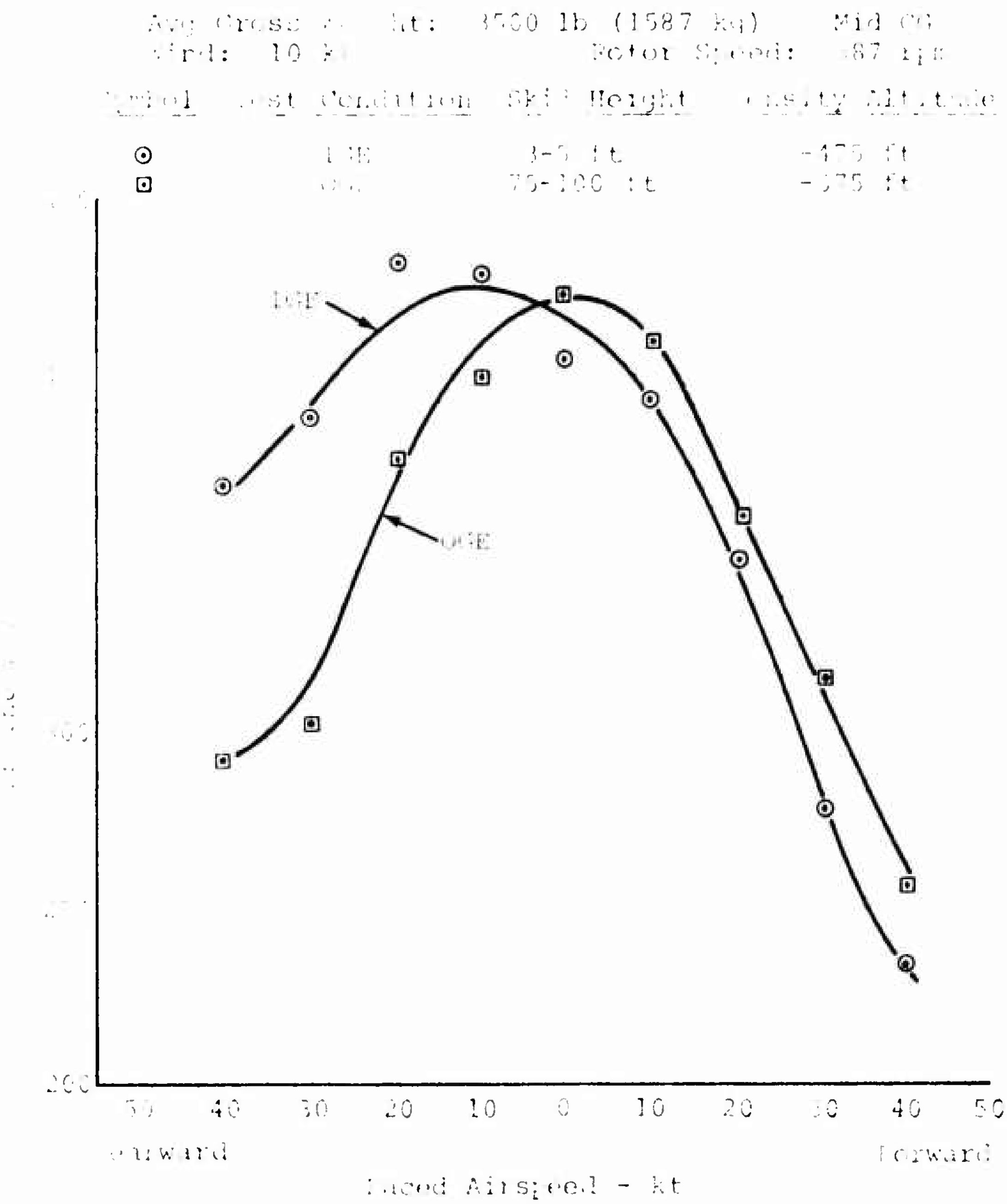


Figure D-2. Power required for forward and rearward flight.

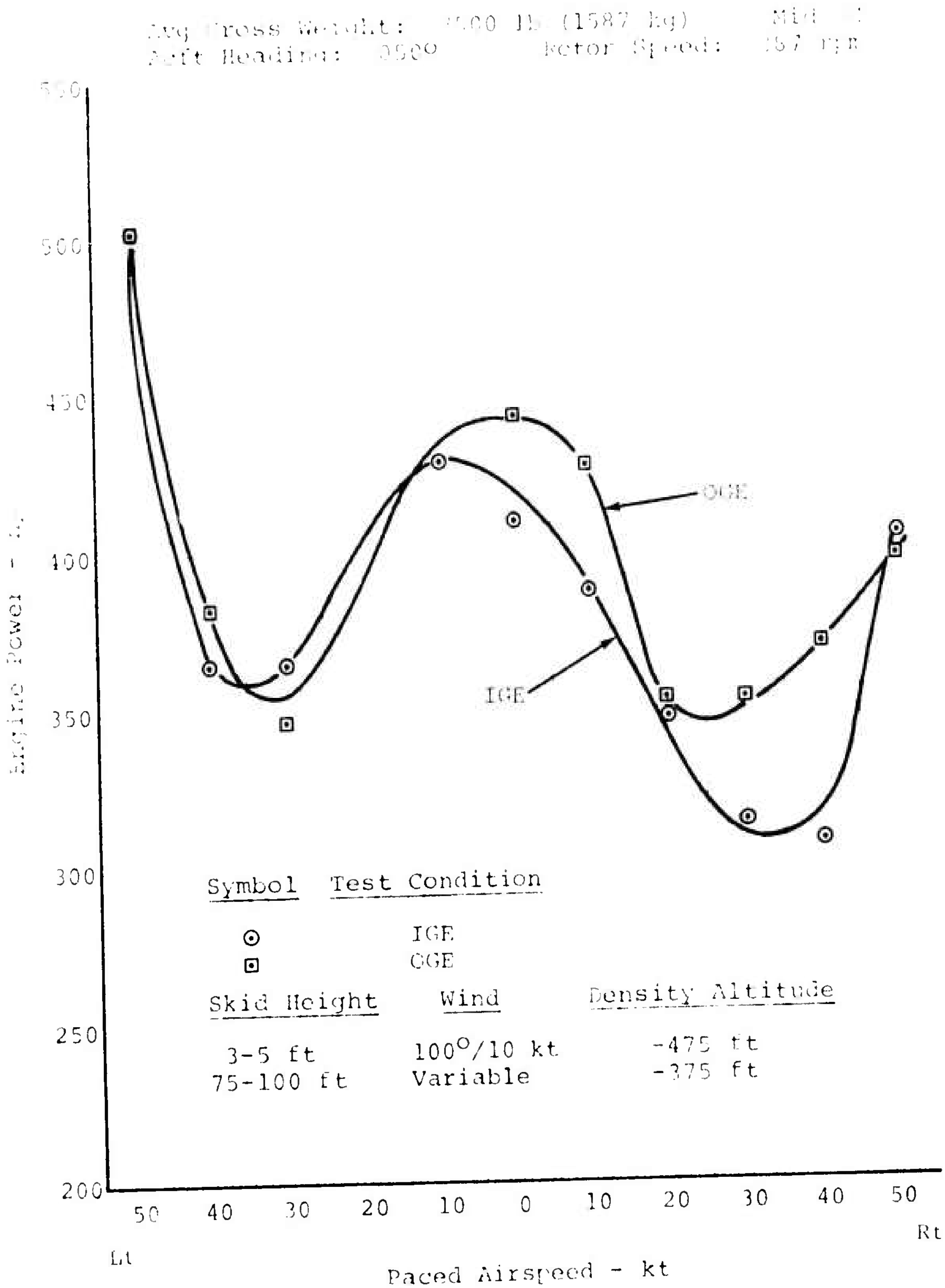


Figure D-3. Power required for sideward flight.

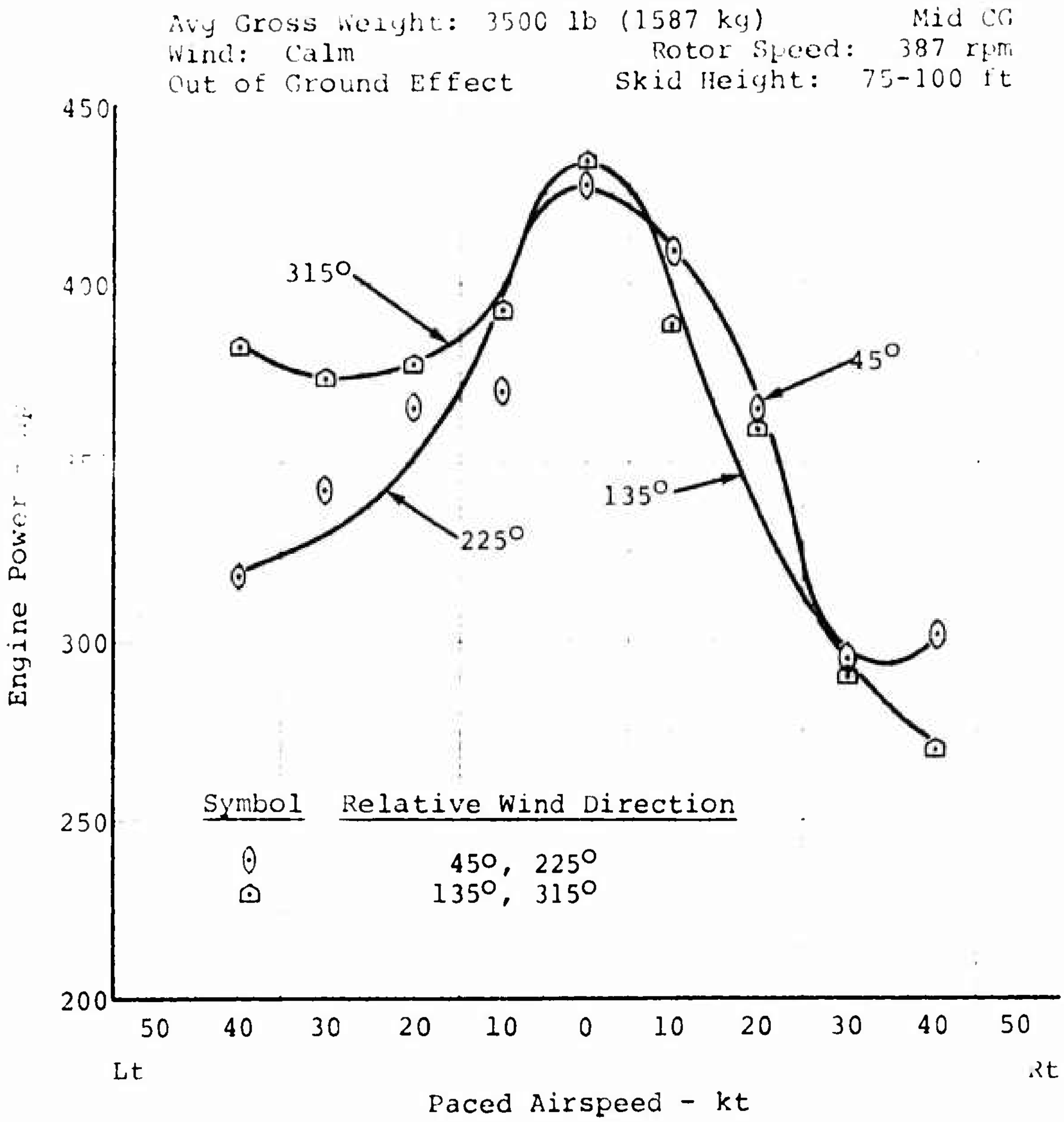


Figure D-4. Power required for 45-degree quartering flight.

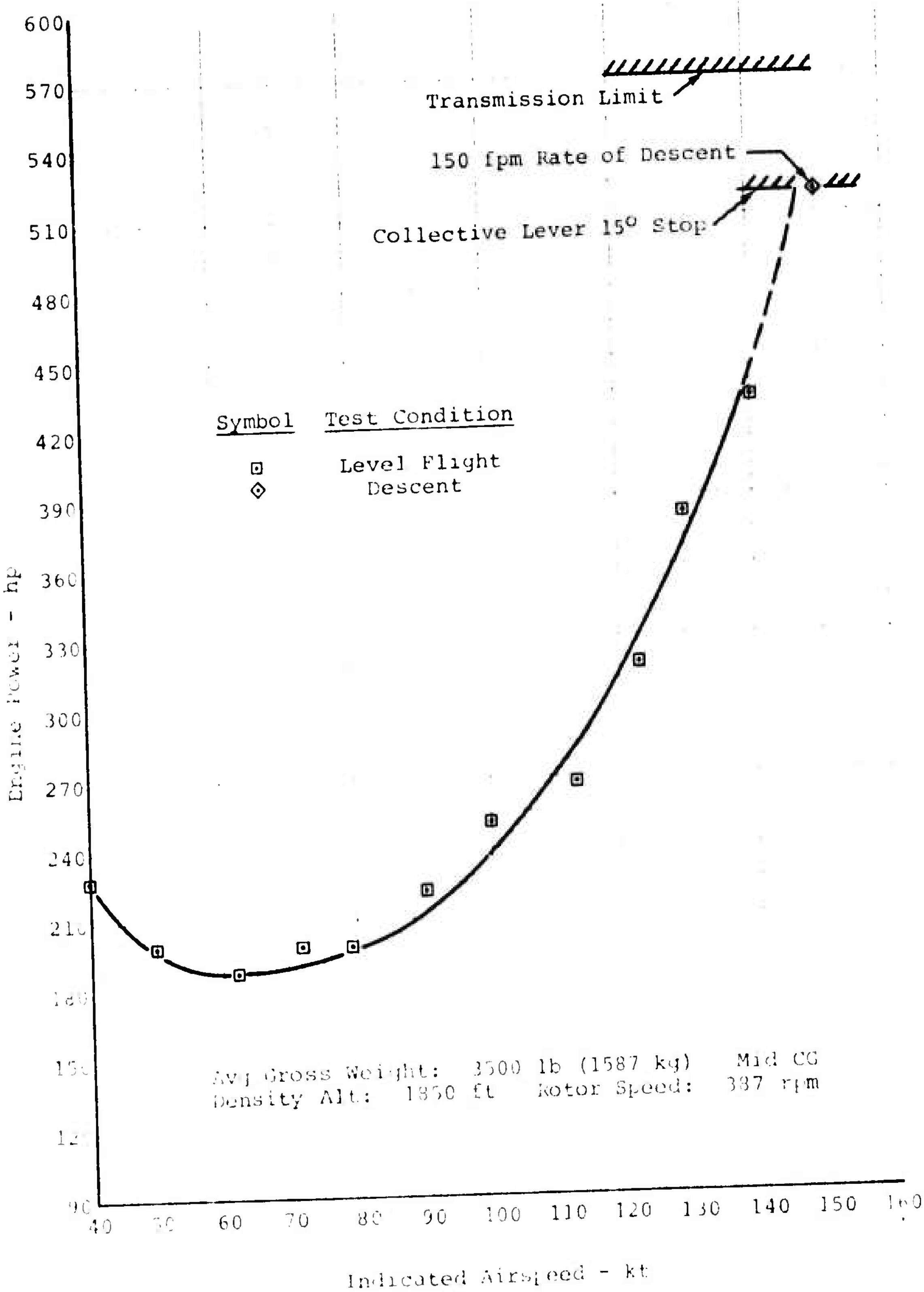


Figure D-5. Power required for level flight.

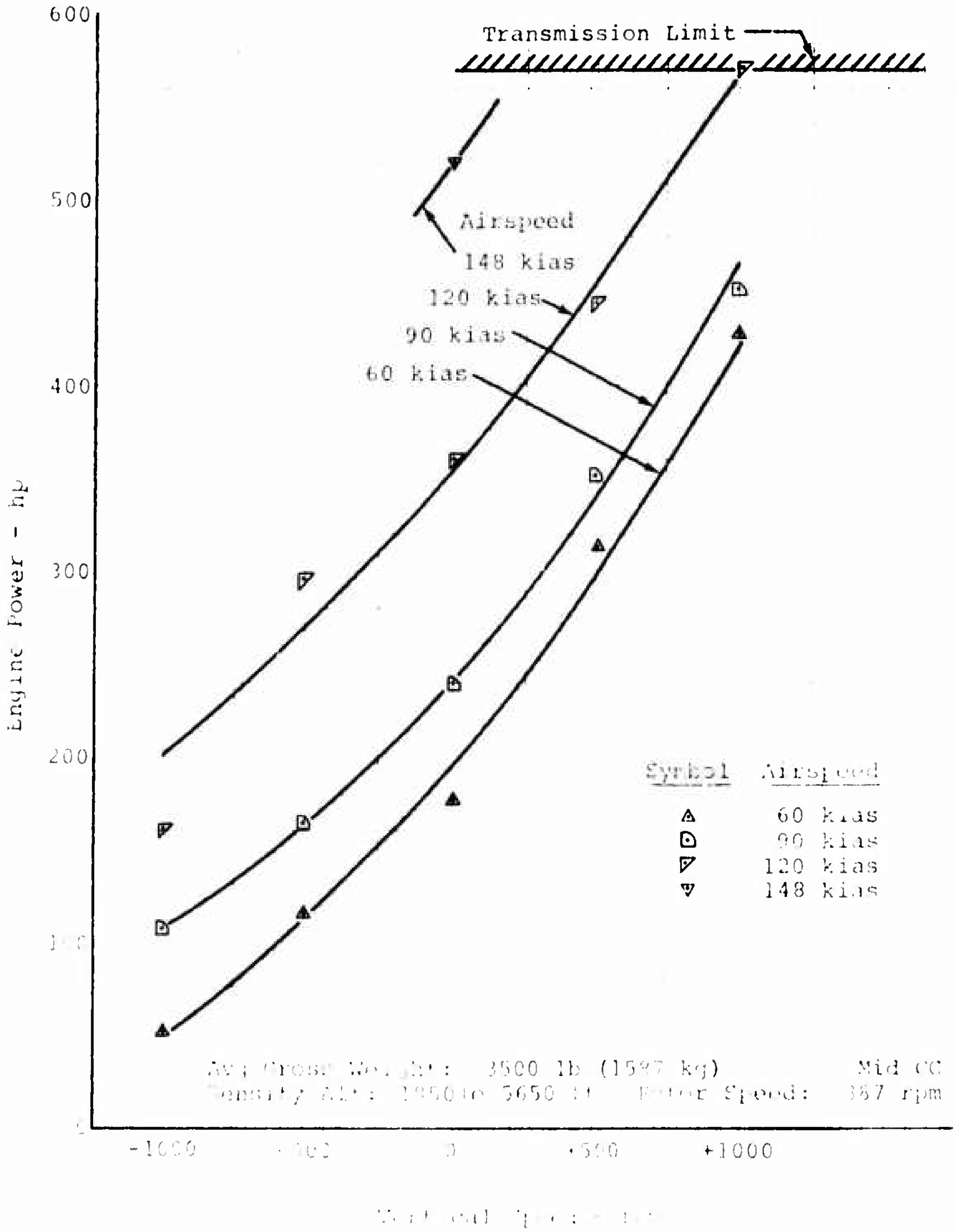


Figure D-6. Power required for ball-centered climb.

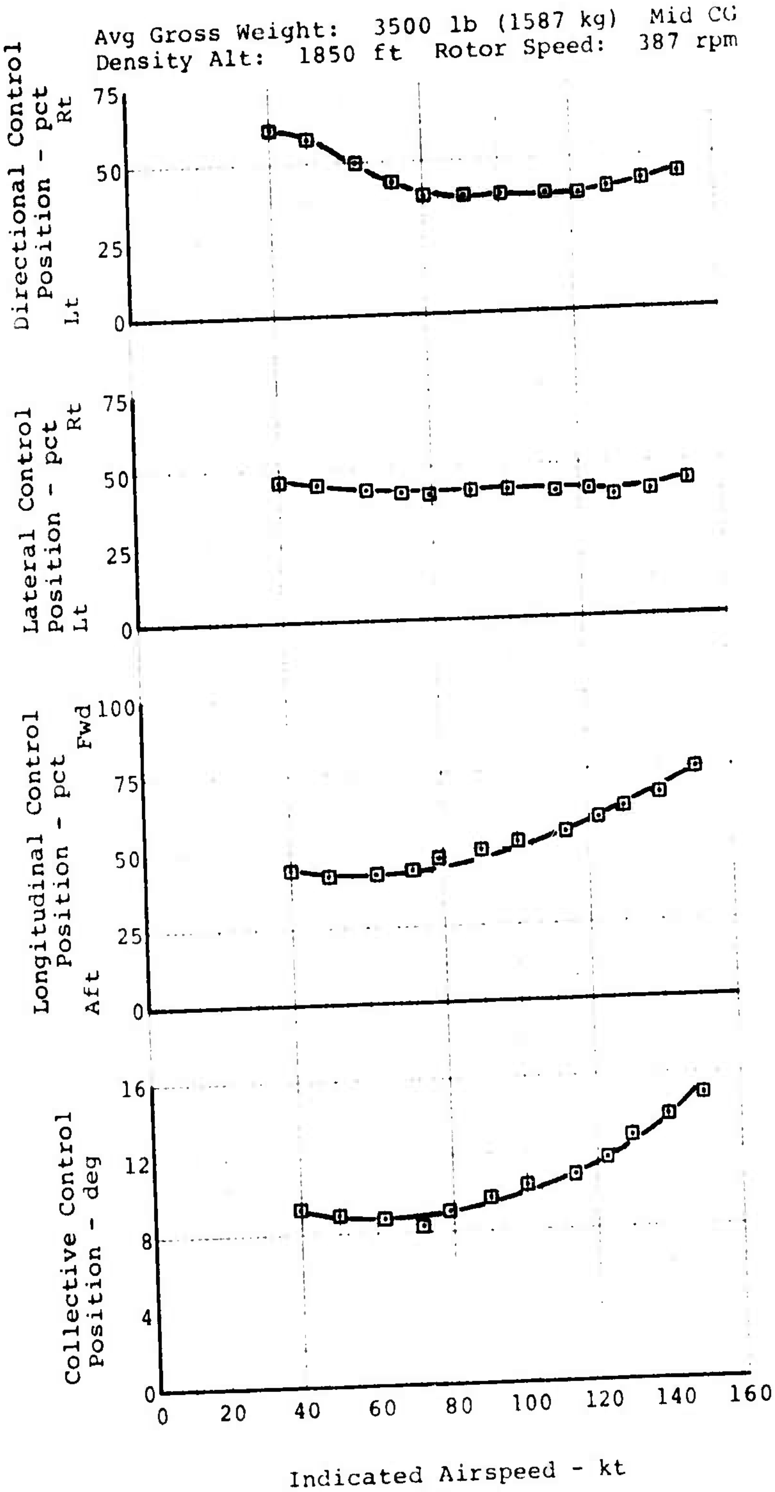


Figure D-7. Control positions in ball-centered forward flight.

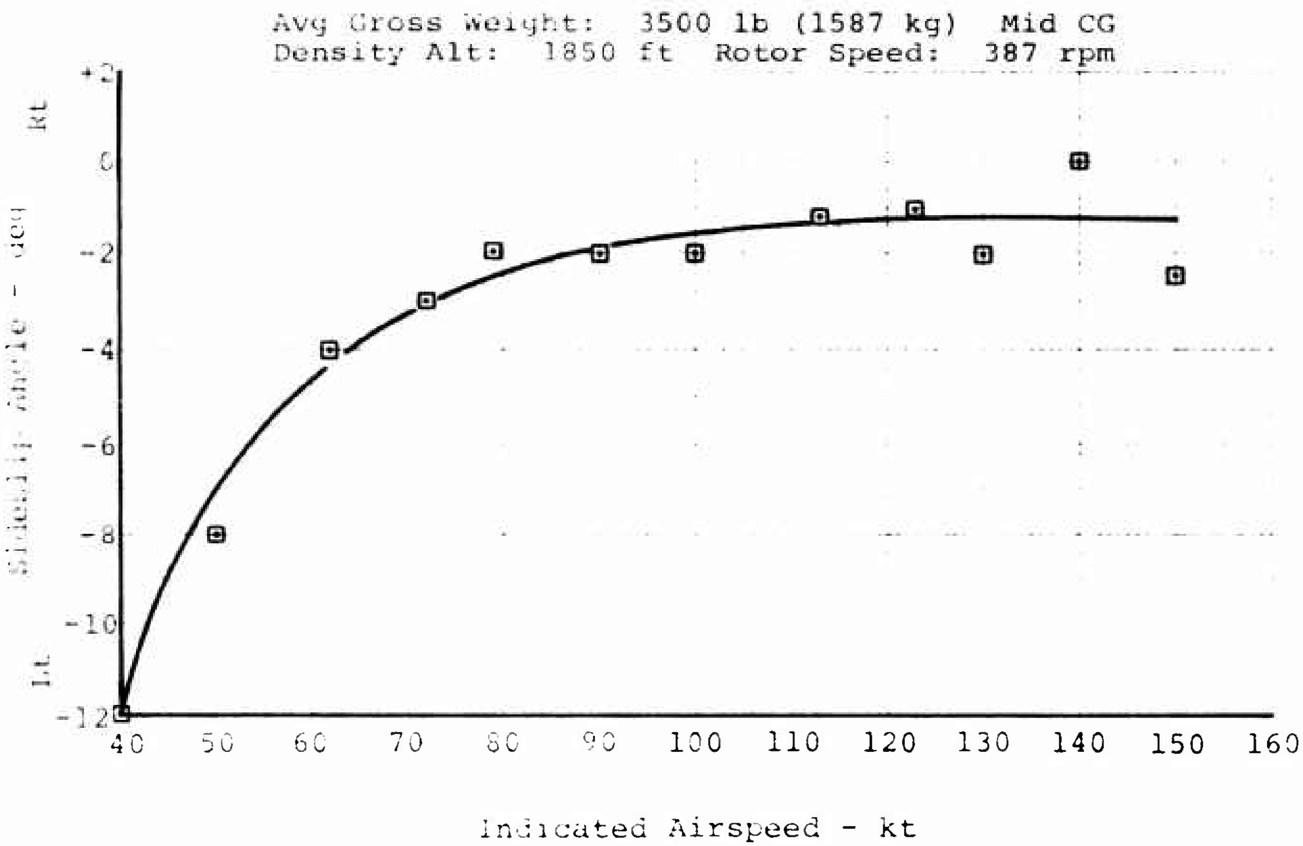


Figure D-8. Sideslip angle for ball-centered forward flight.

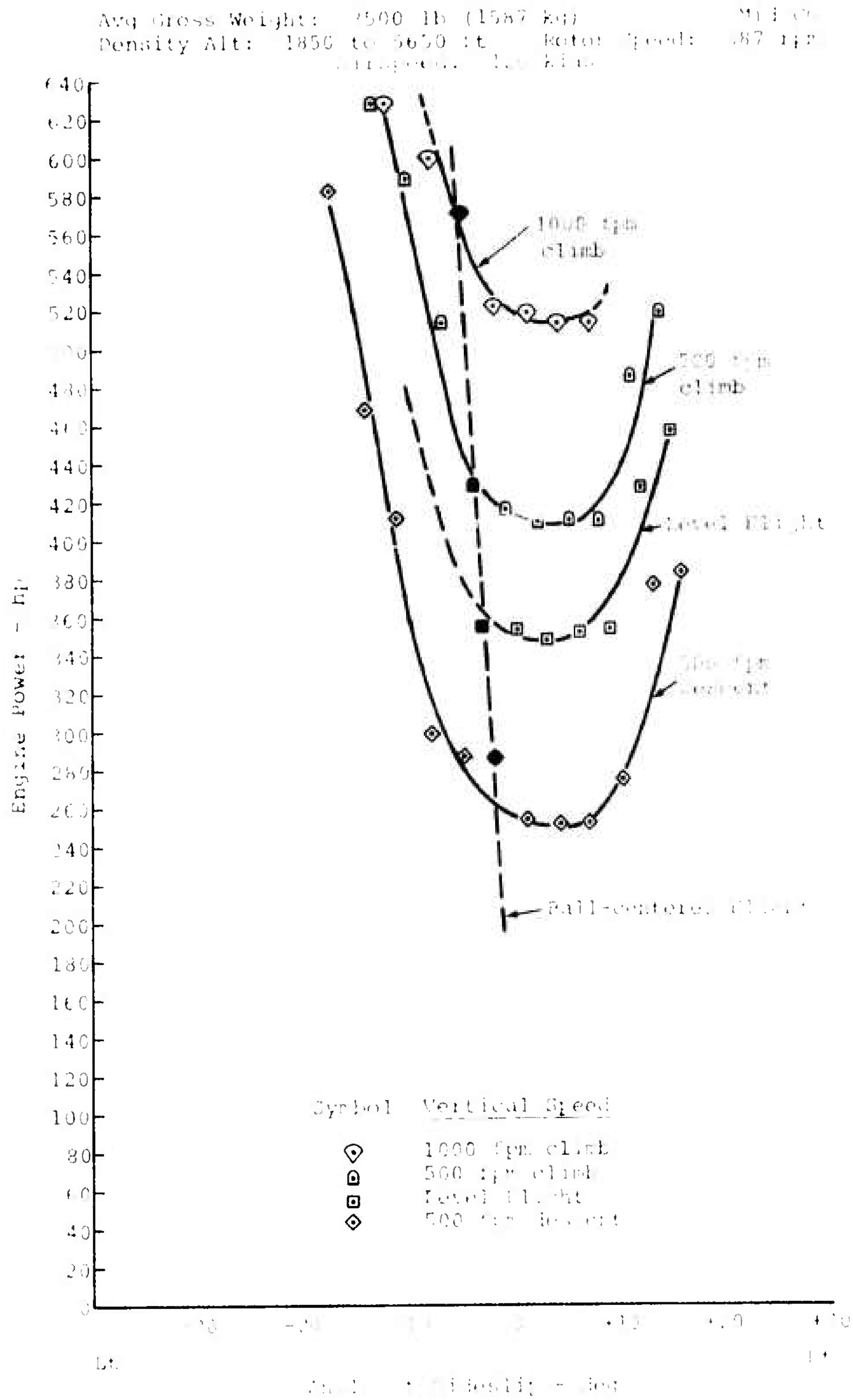


Figure D-9. Power required in steady heading sideslips.

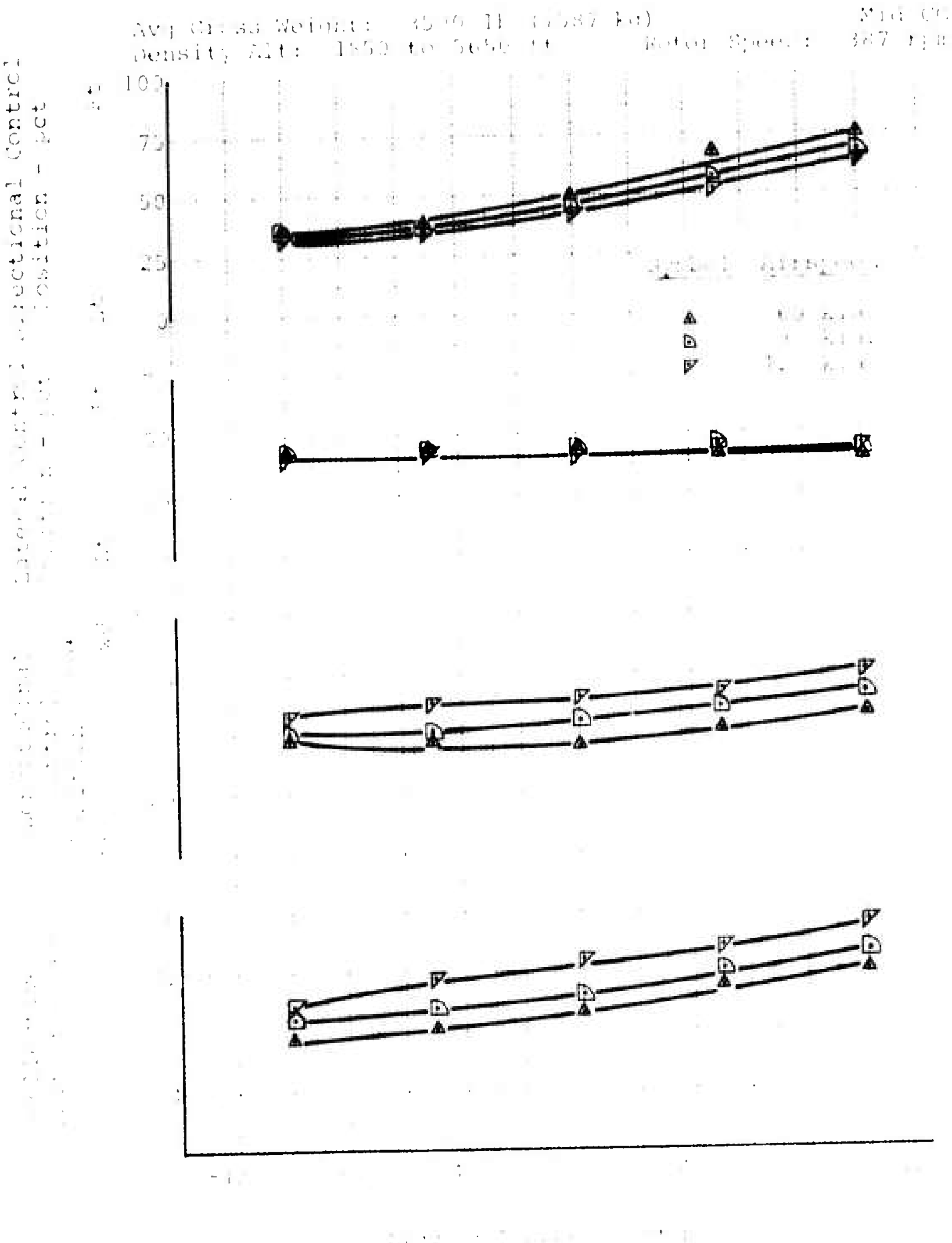


Figure D-10. Control position variation with vertical speed.

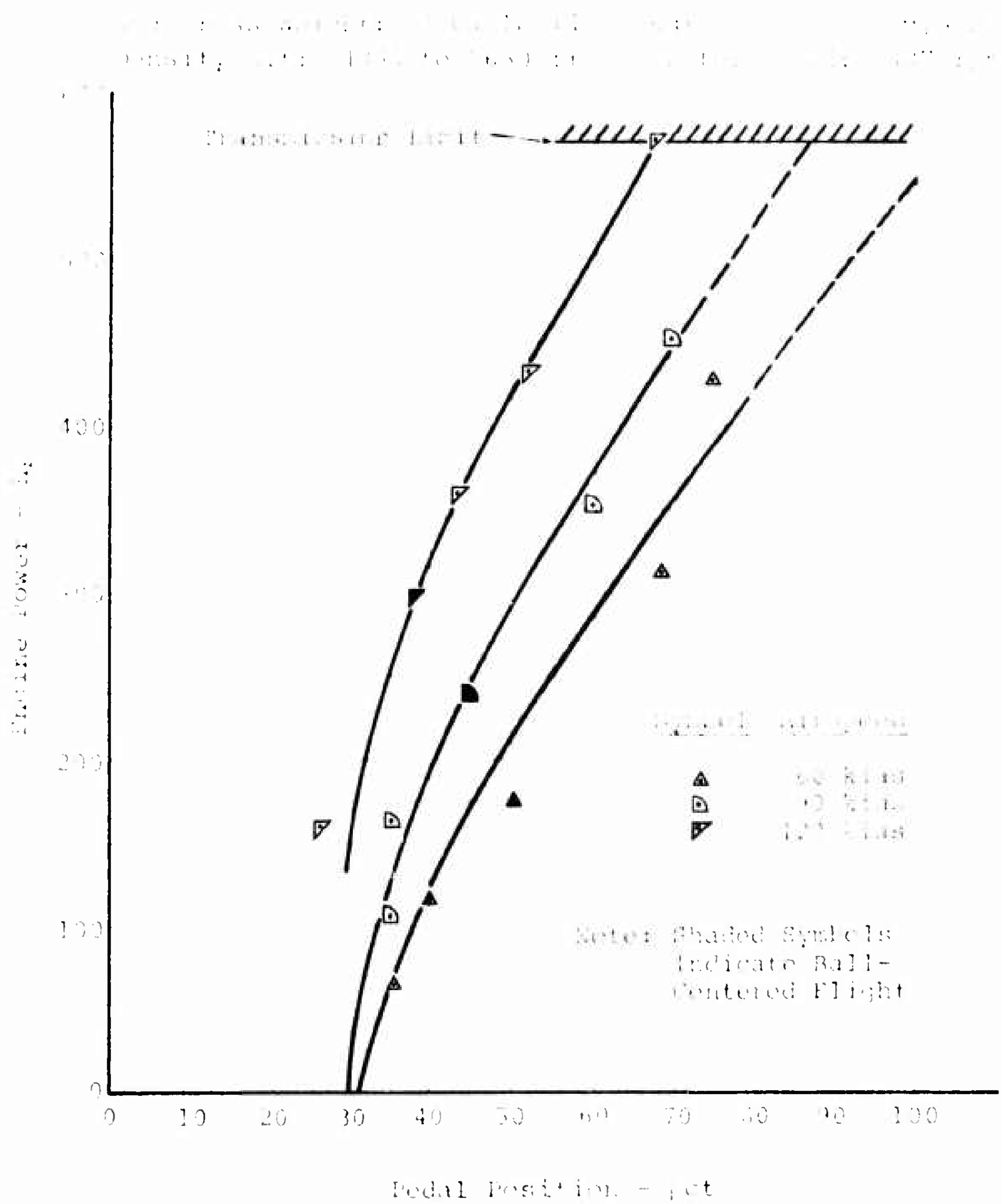


Figure D-11. Variation of pedal position with engine power.

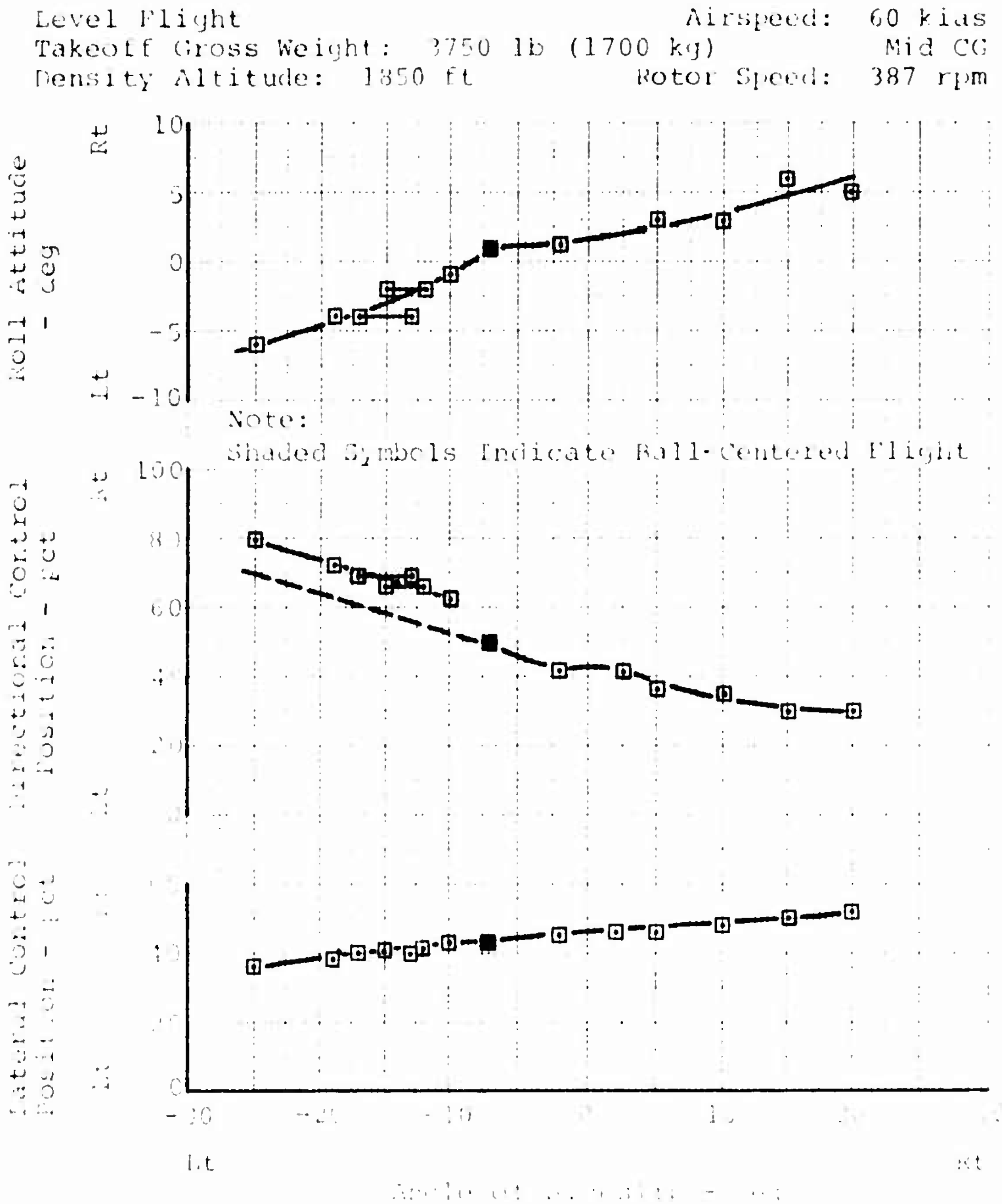


Figure D-12. Static lateral-directional stability.

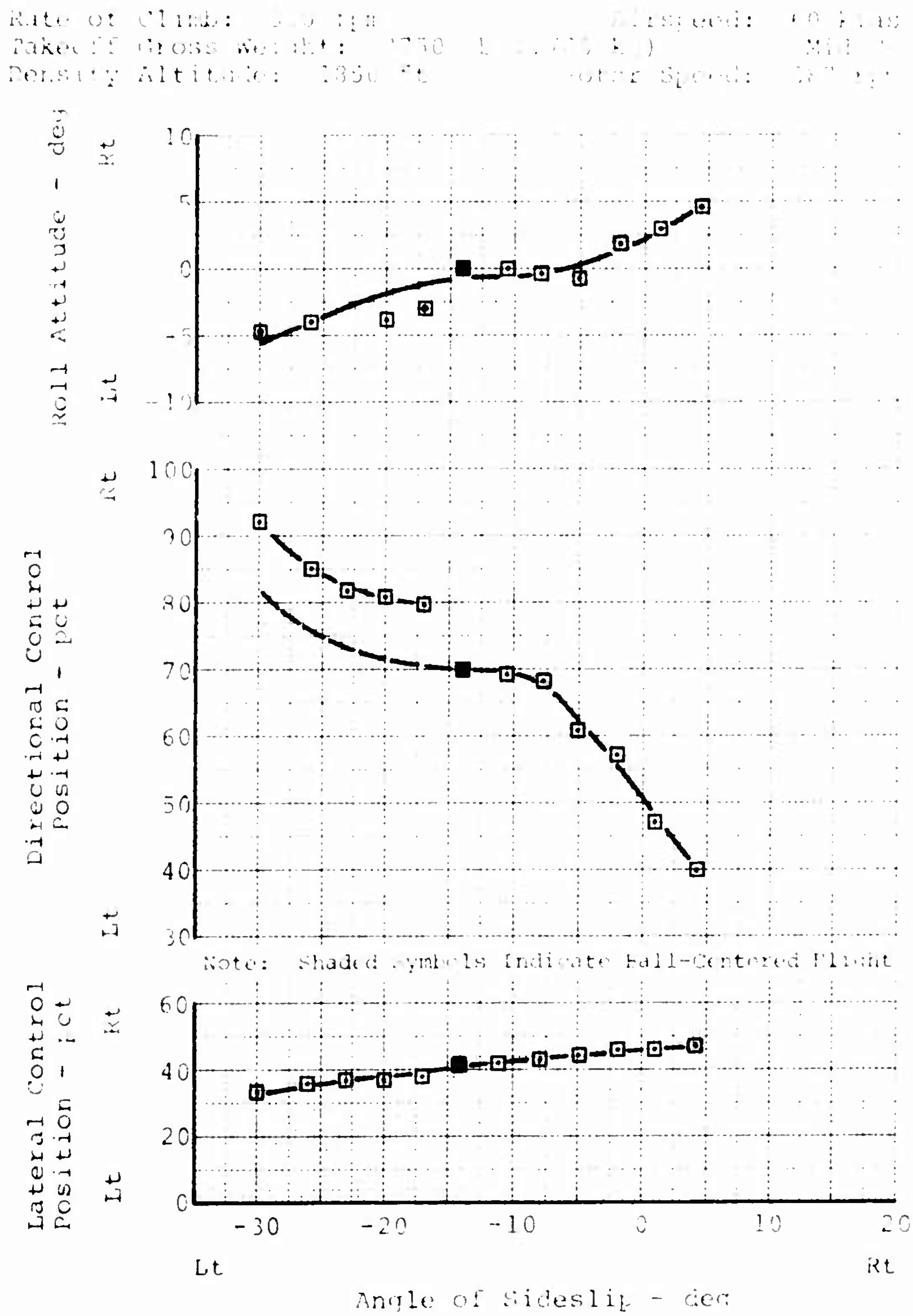


Figure D-13. Static lateral-directional stability.

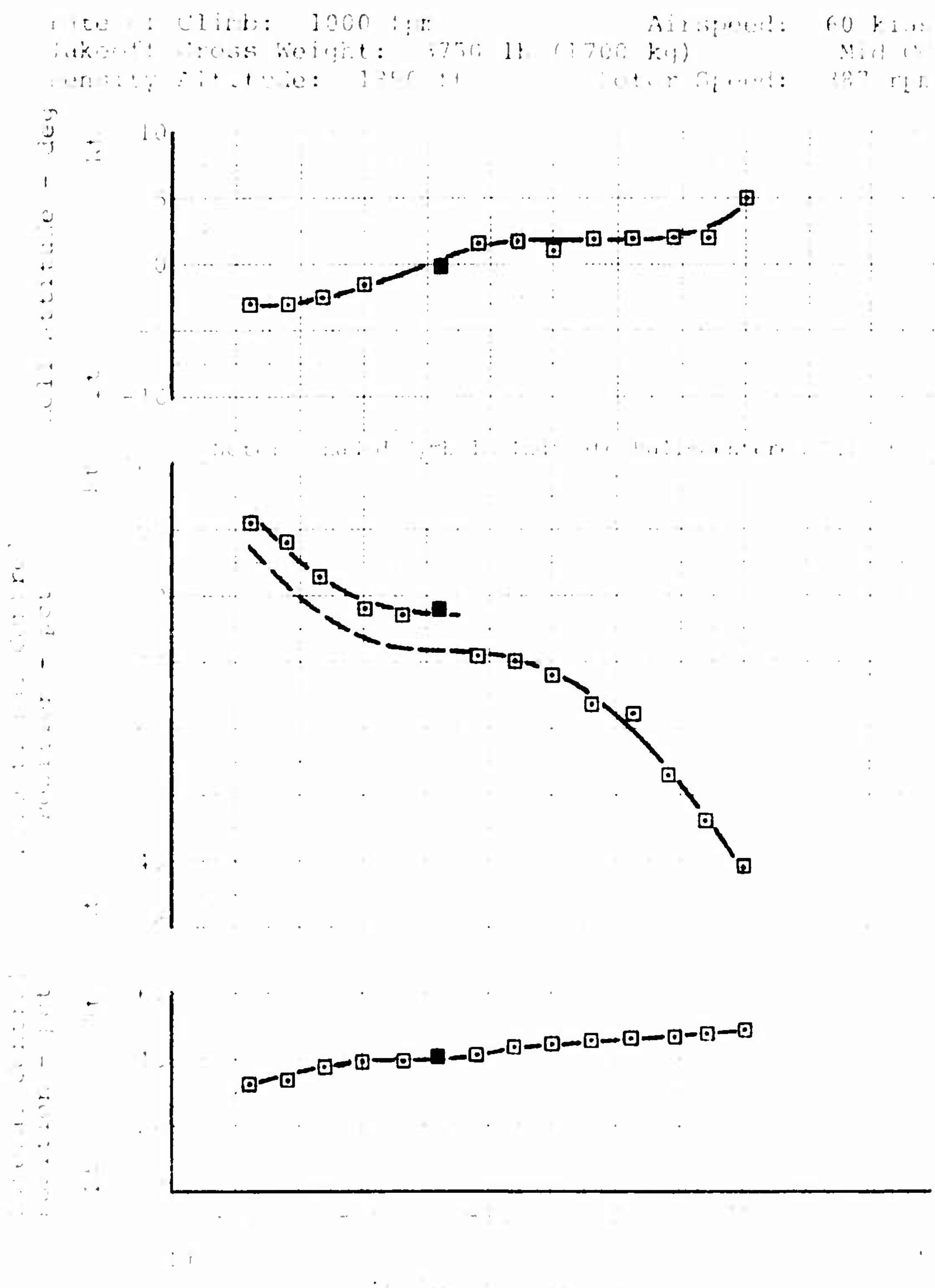


Figure D-14. Static lateral-directional stability.

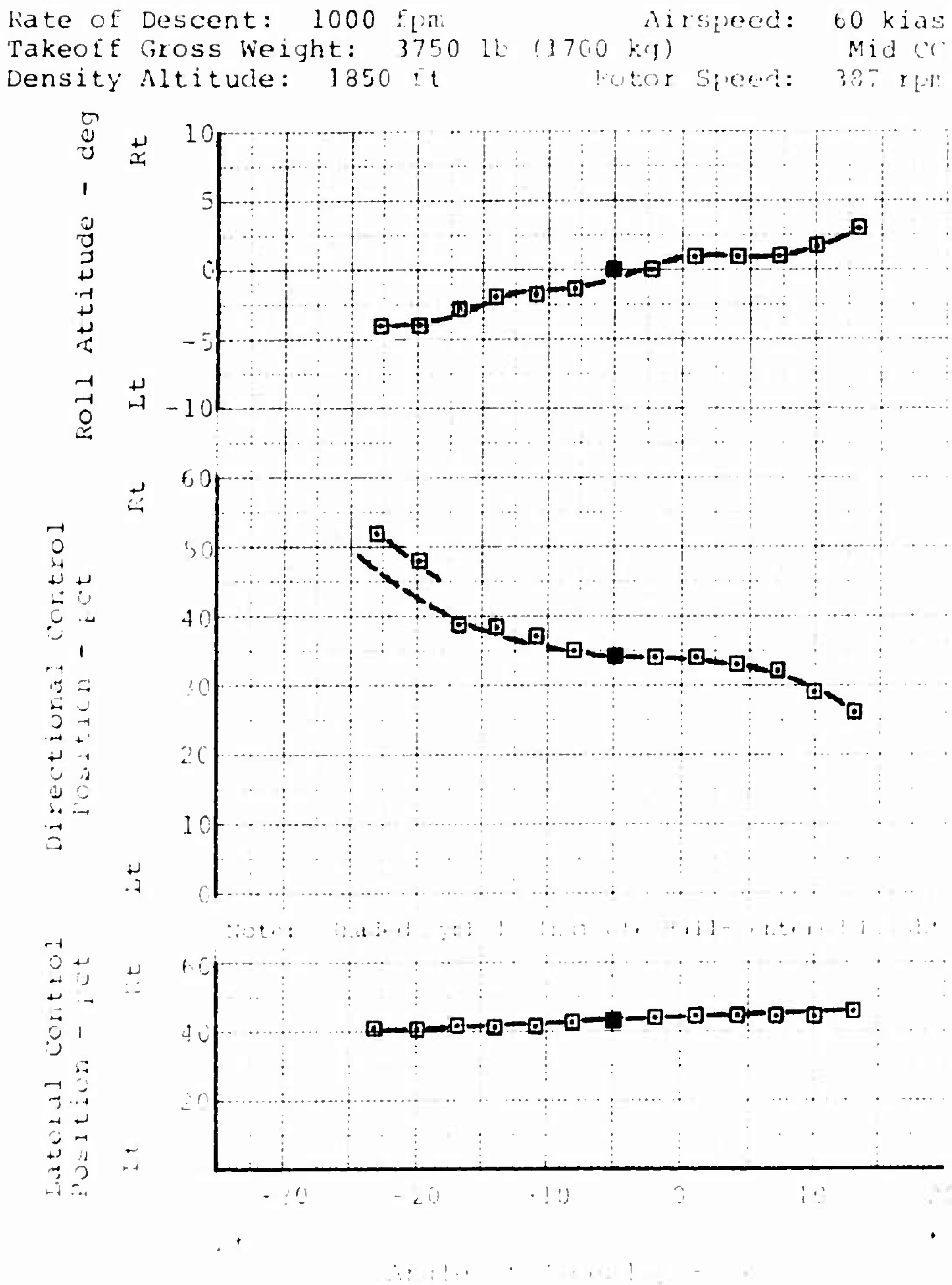


Figure D-15. Static lateral-directional stability.

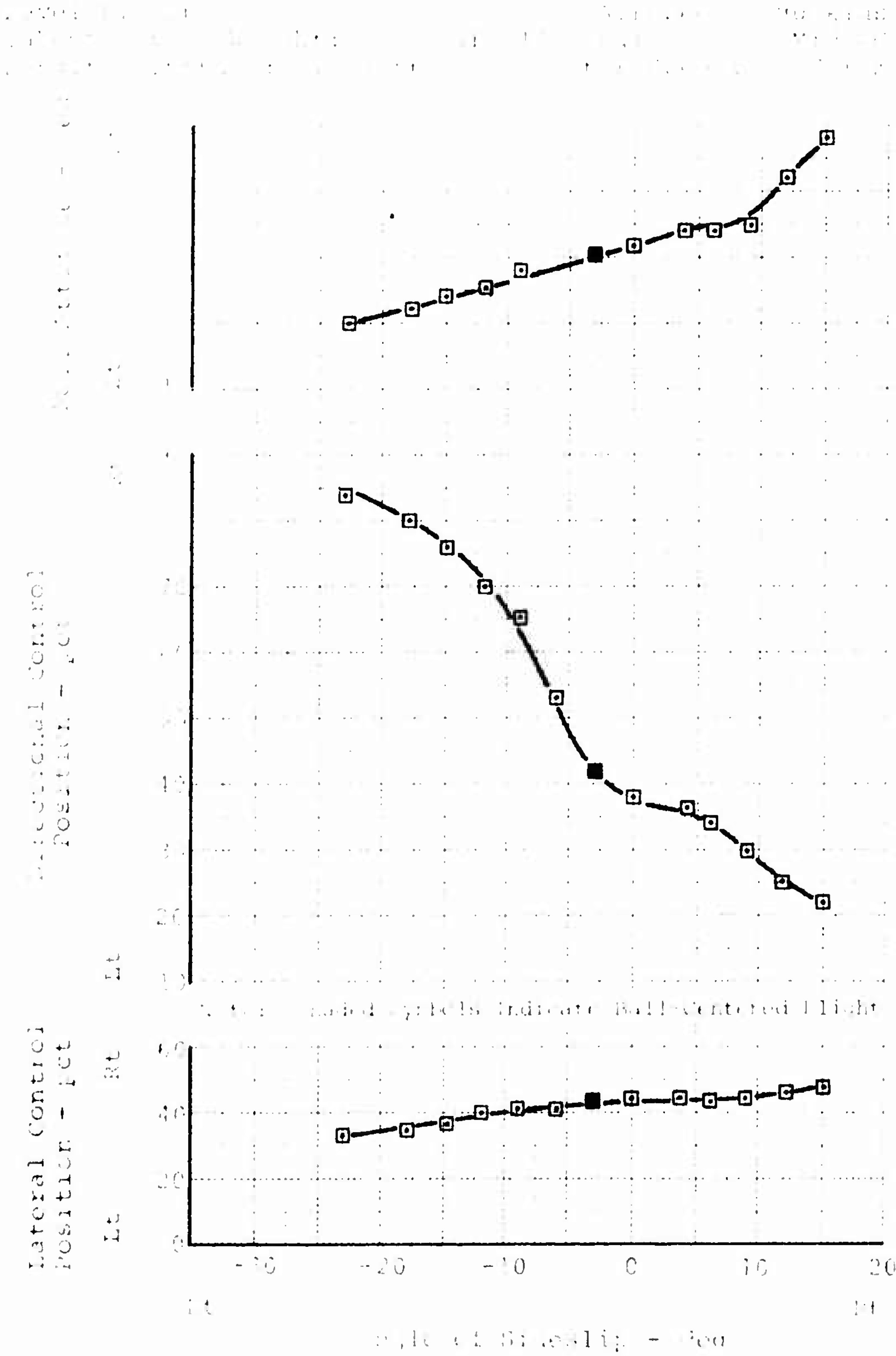


Figure D-16. Static lateral-directional stability.

Rate of Climb: 500 fpm Airspeed: 90 kias
Takeoff Gross Weight: 3750 lb (1700 kg) Mid CG
Density Altitude: 1850 ft Rotor Speed: 387 rpm

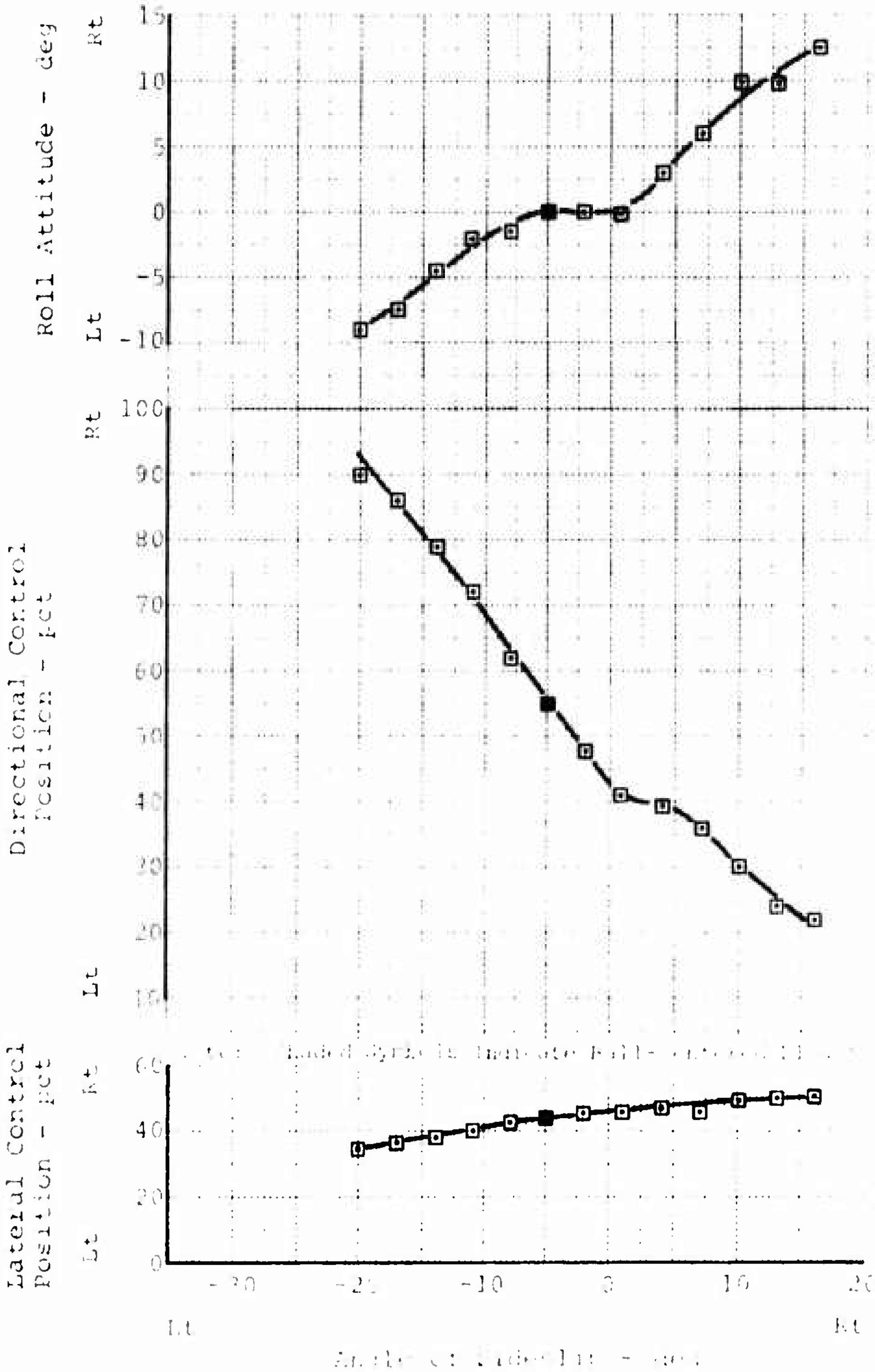


Figure D-17. Static lateral-directional stability.

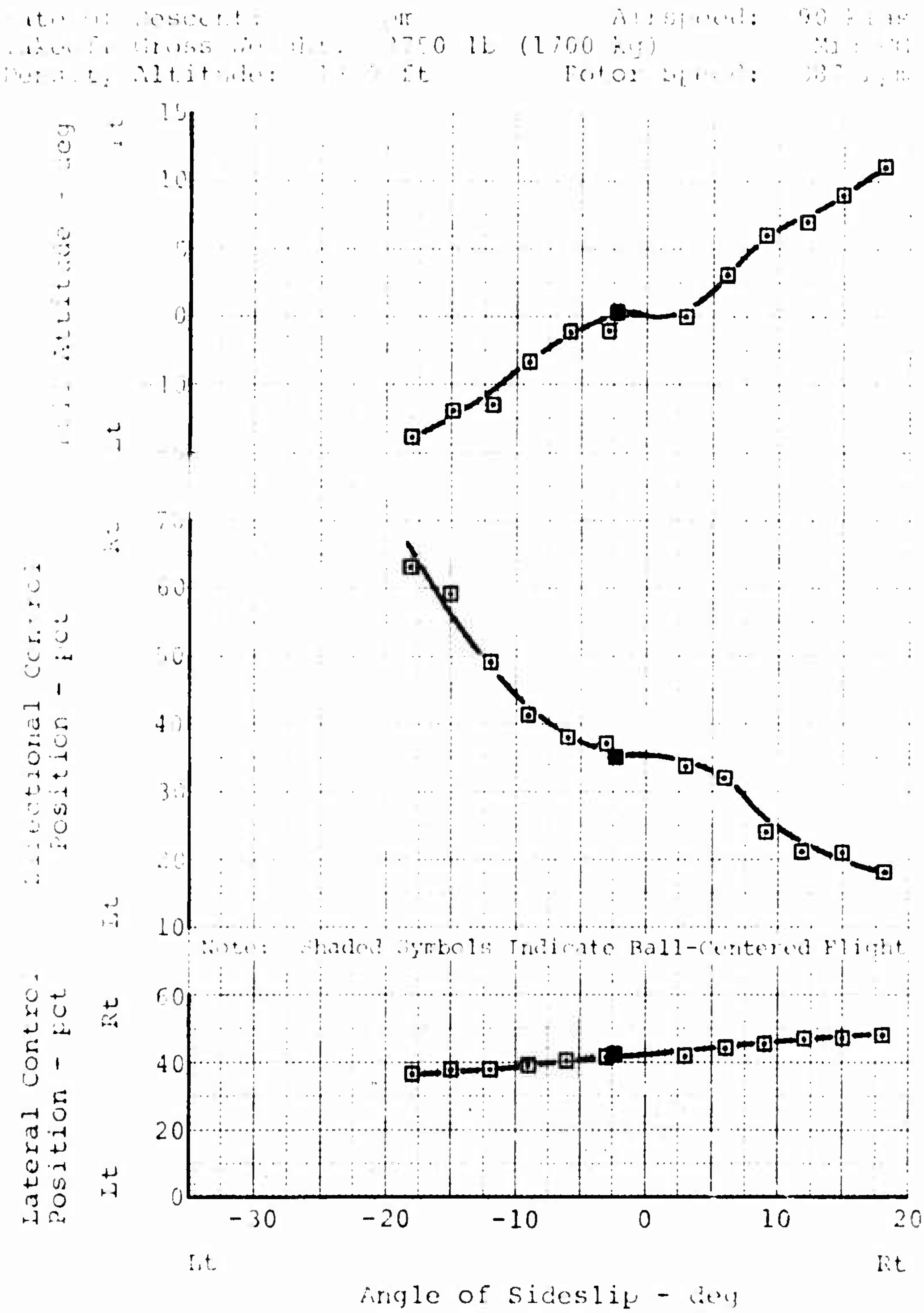


Figure D-18. Static lateral-directional stability.

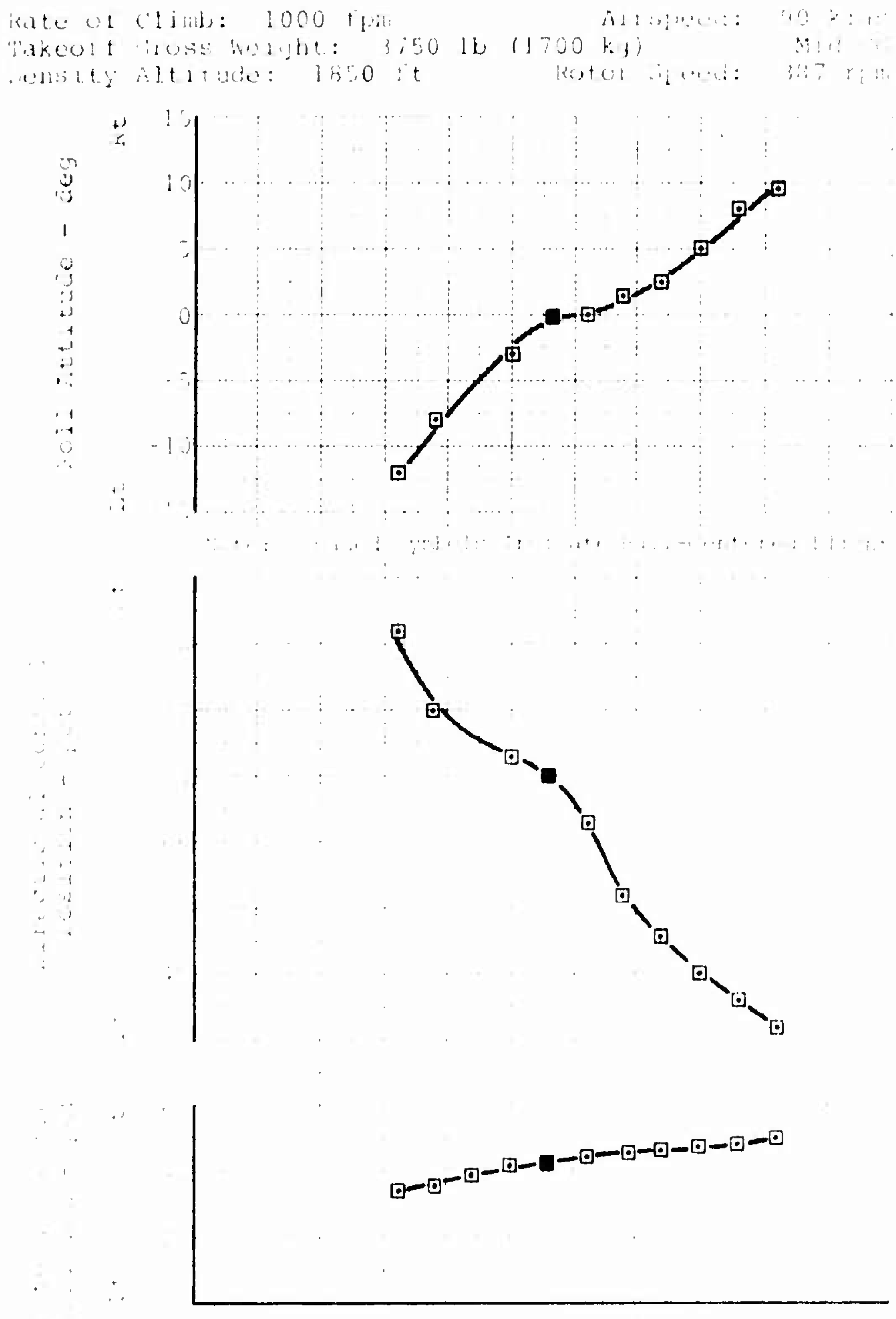


Figure D-19. Static lateral-directional stability.

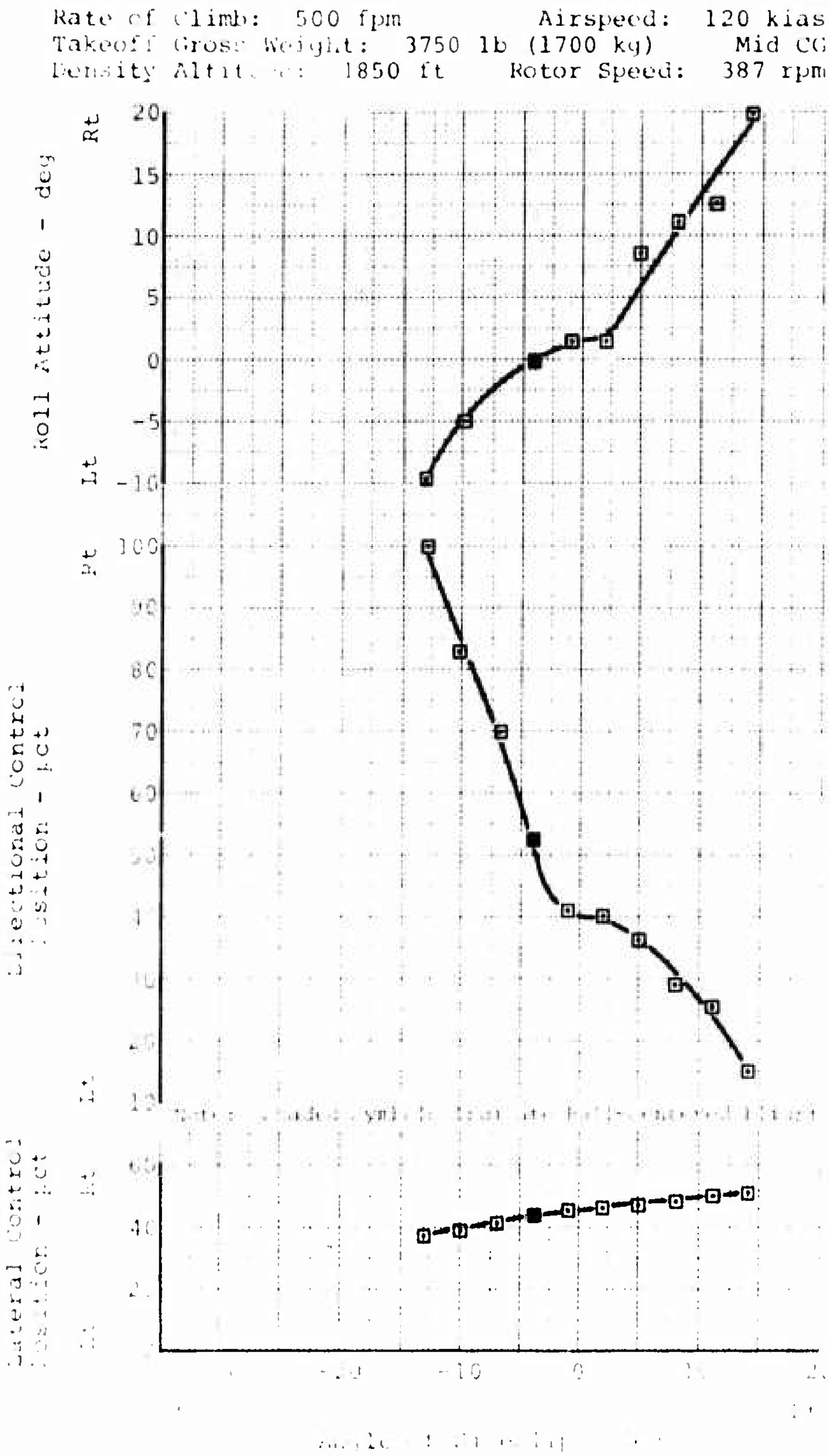


Figure D-20. Static lateral-directional stability.

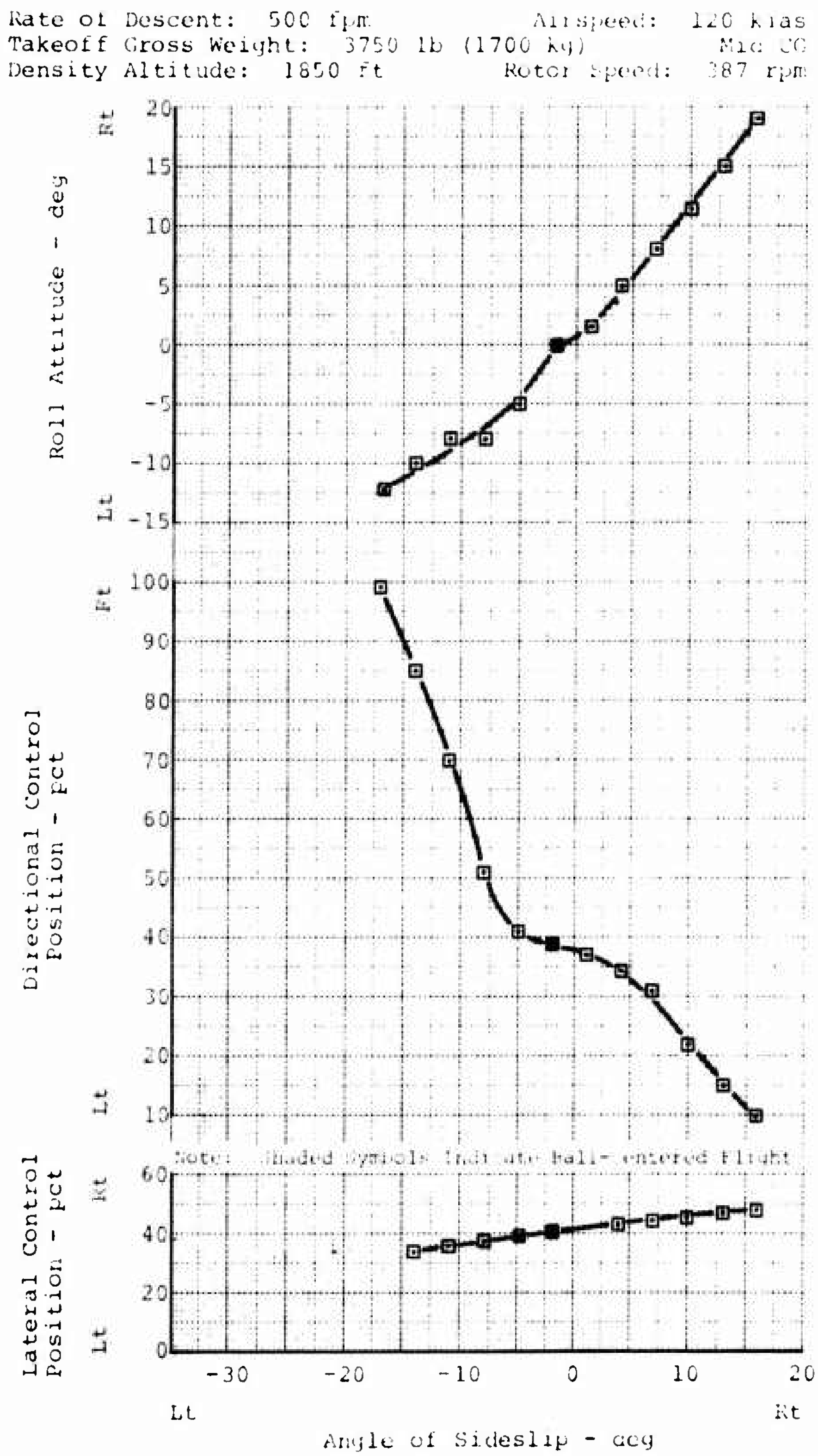


Figure D-21. Static lateral-directional stability.

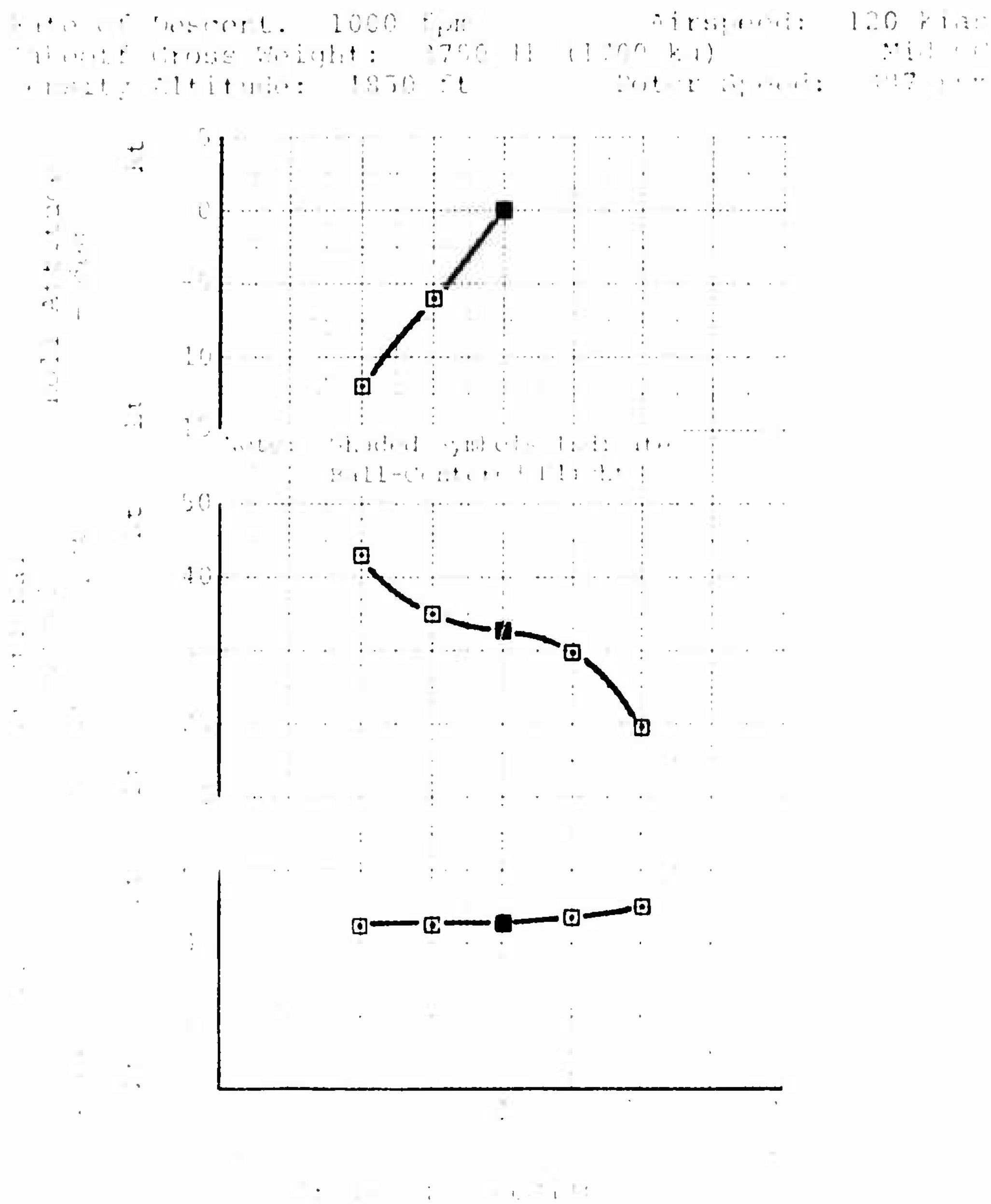


Figure D-22. Static lateral-directional stability.

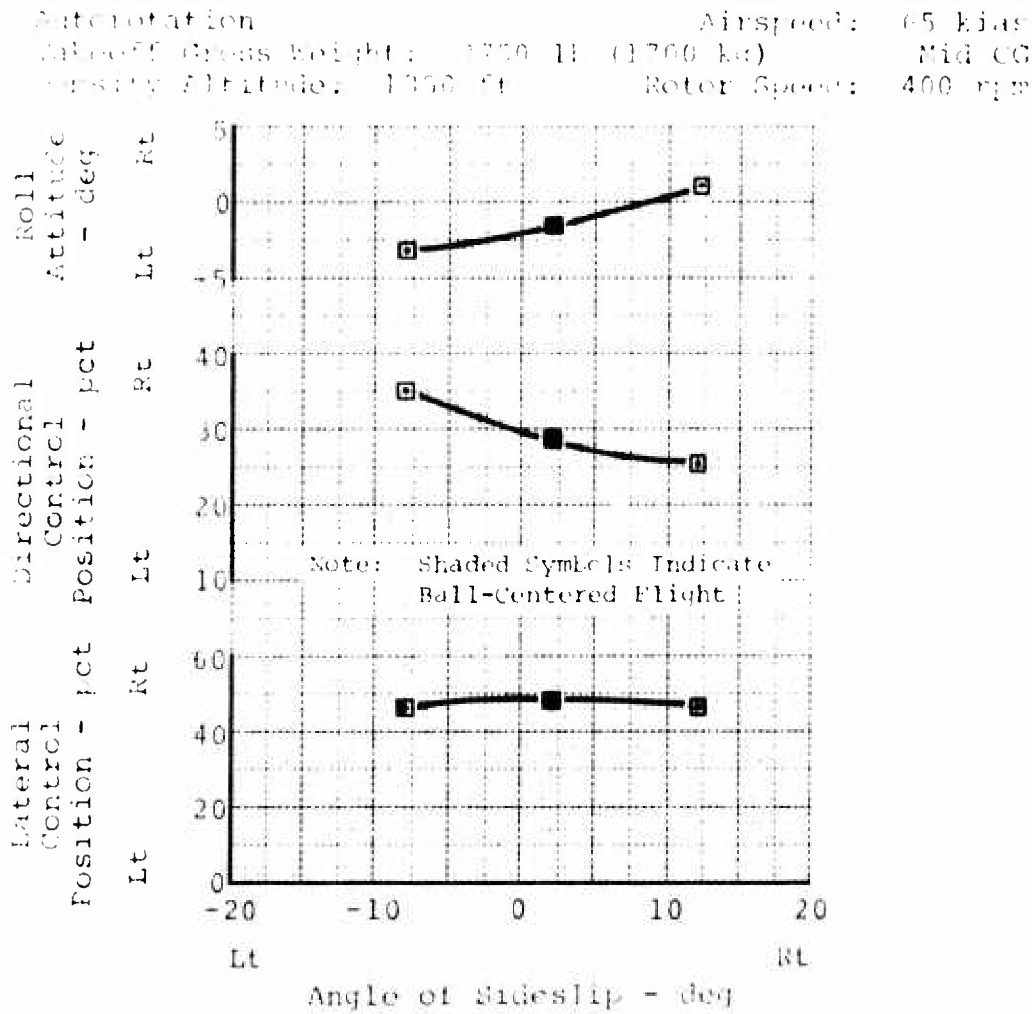


Figure D-23. Static lateral-directional stability.

Autorotation Airspeed: 90 kias
Takeoff Gross Weight: 3750 lb (1700 kg) Mid CG
Density Altitude: 1850 ft Rotor Speed: 400 rpm

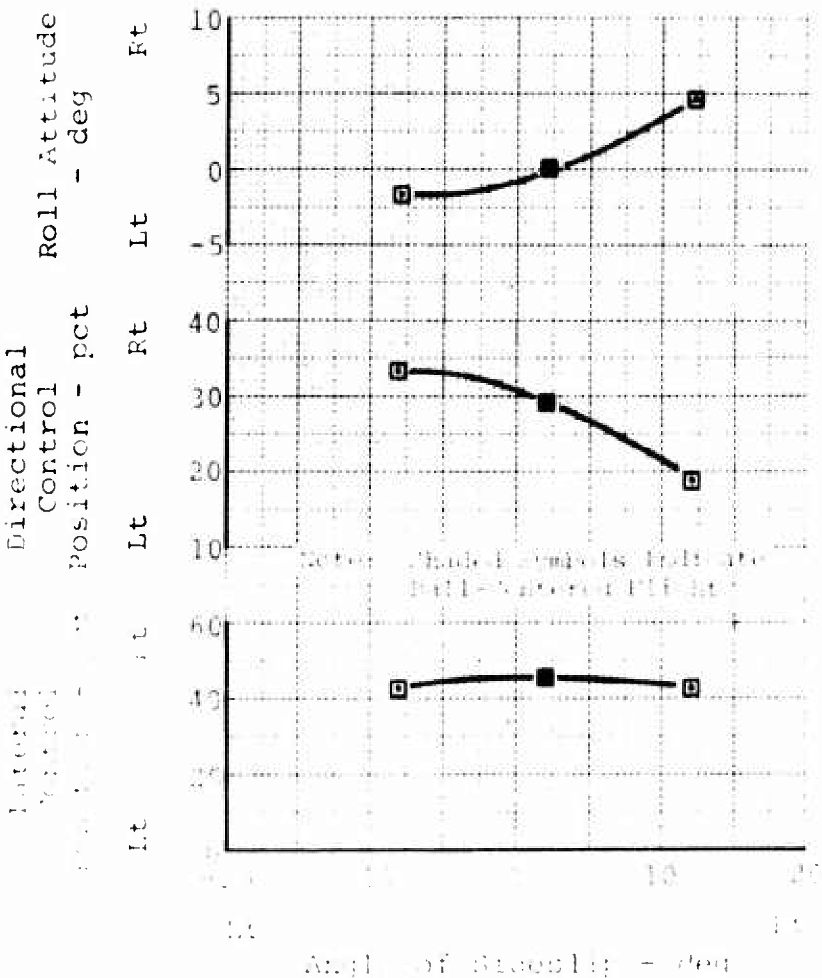


Figure D-24. Static lateral-directional stability.

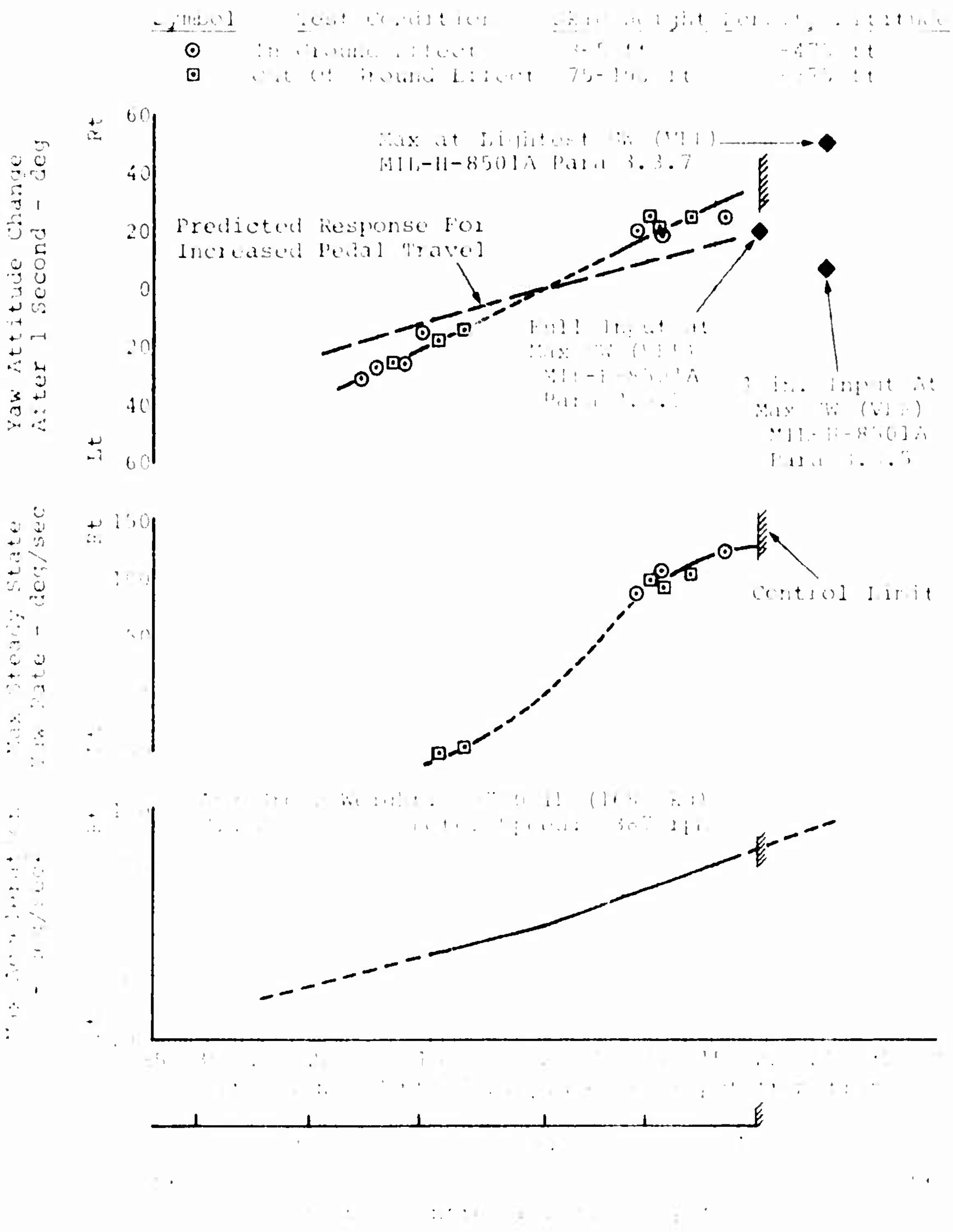


Figure D-25. Directional control response and sensitivity.

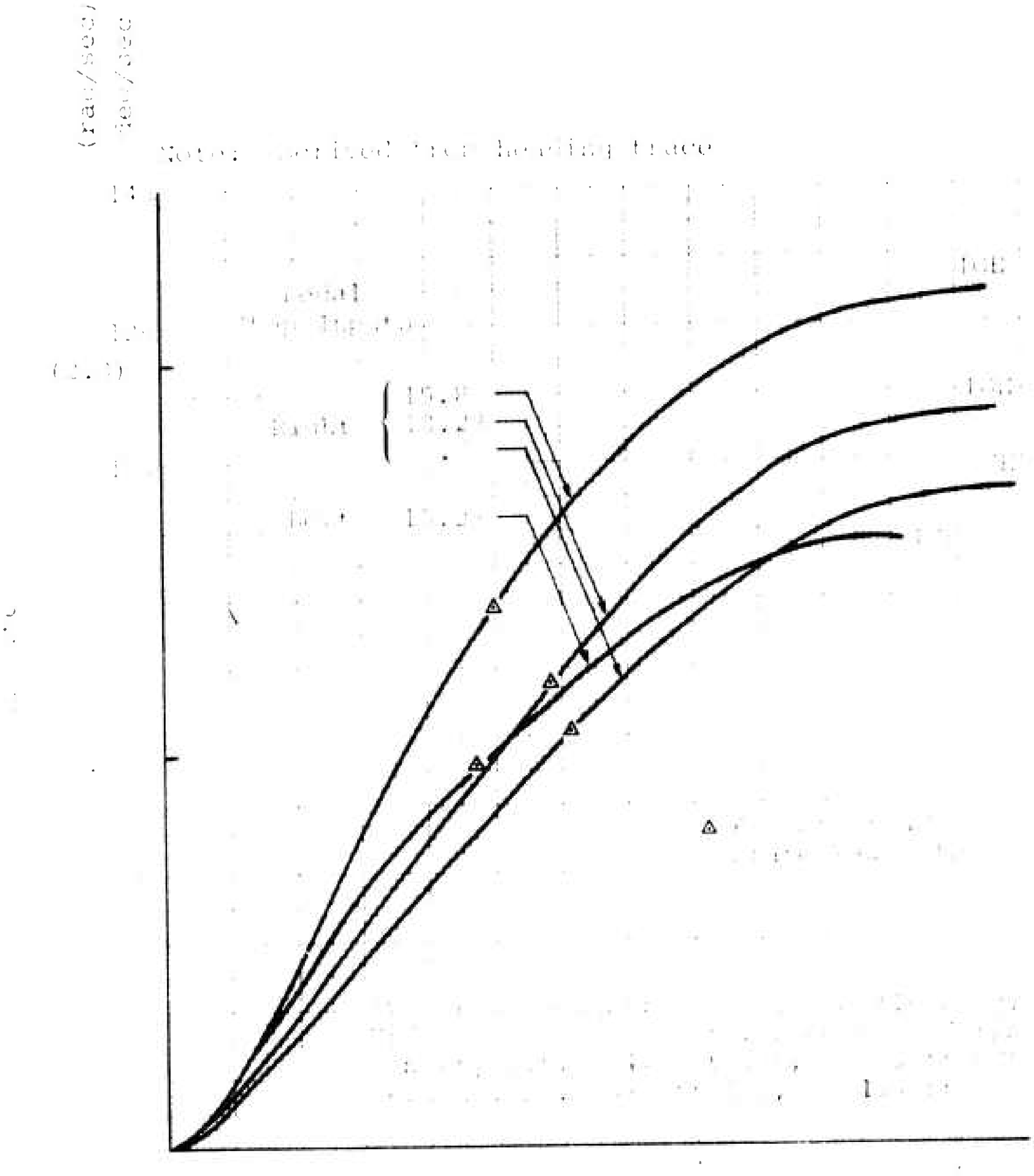


Figure D-26. Yaw rate response to directional control step input.

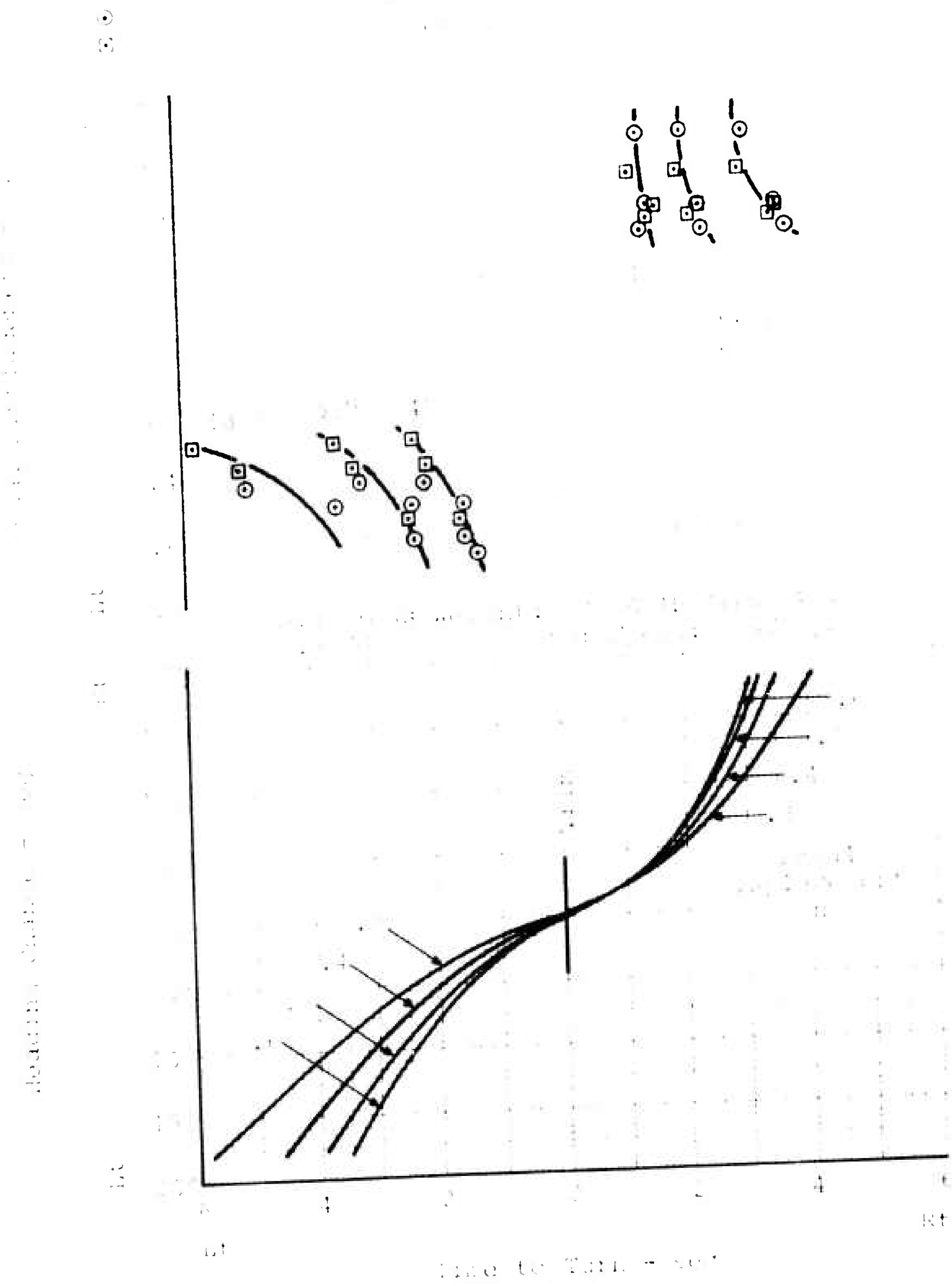


Figure D-27. Time required to perform hovering turns.

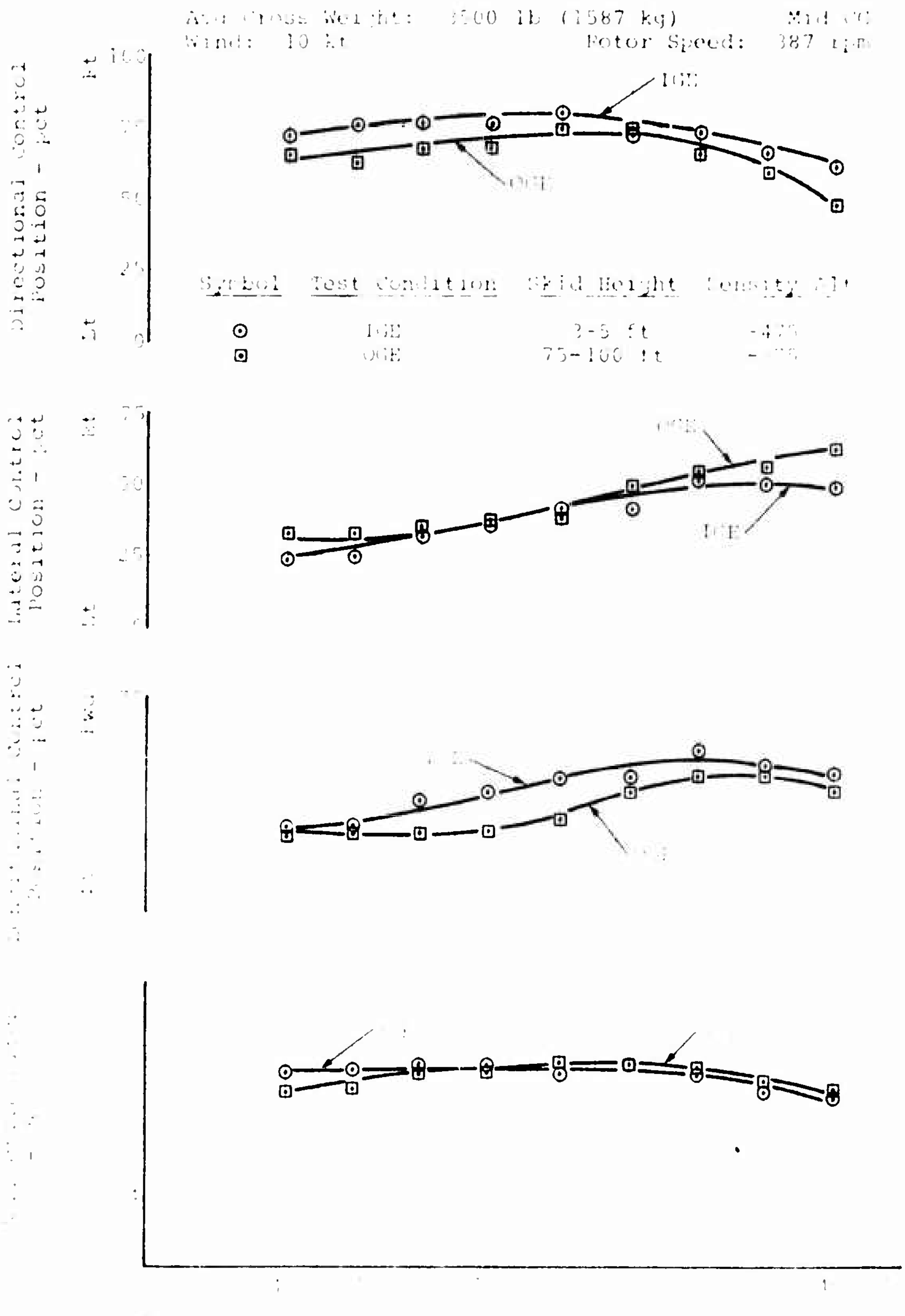


Figure D-28. Control positions for forward and rearward flight.

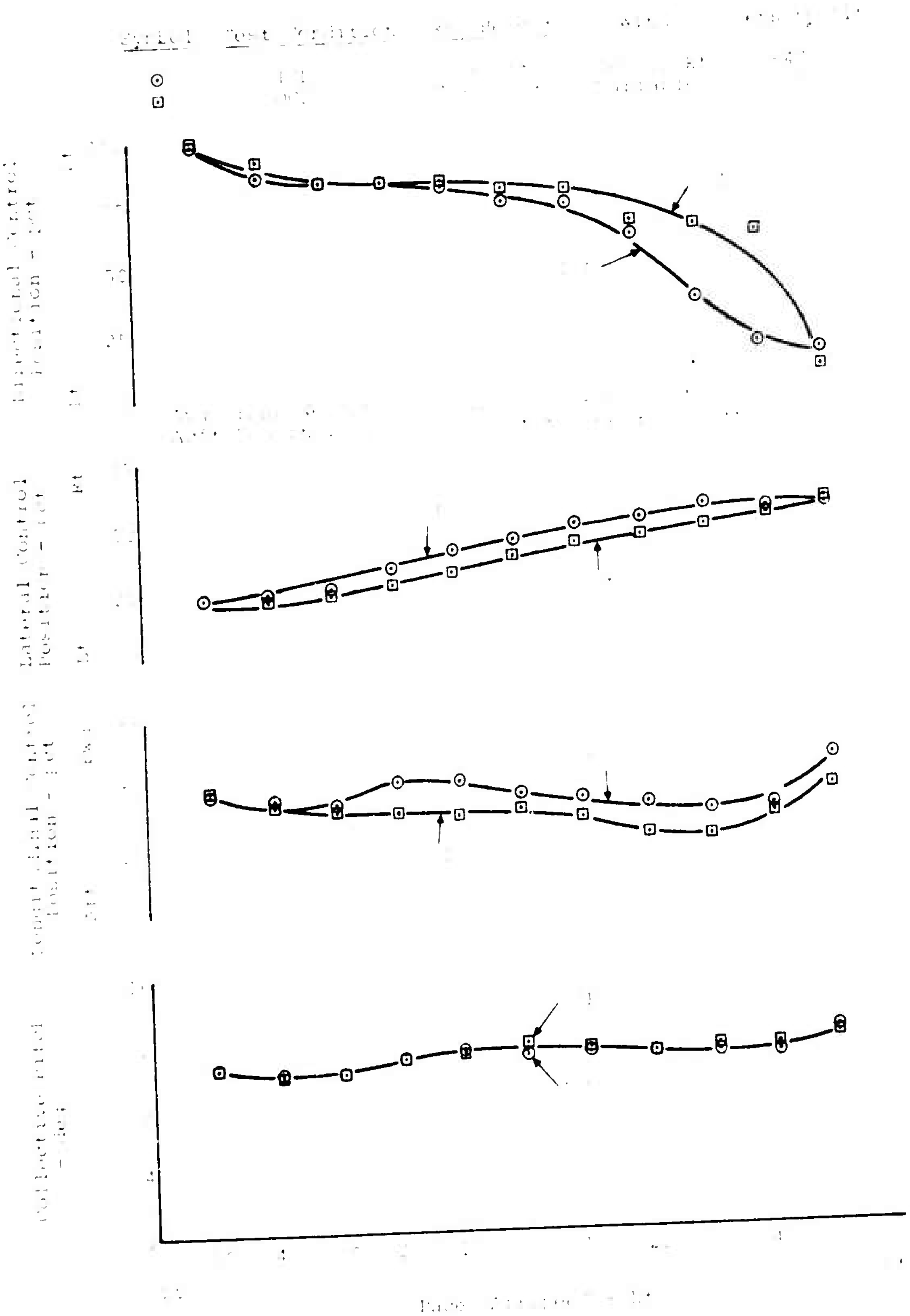


Figure D-29. Control positions for sideward flight.

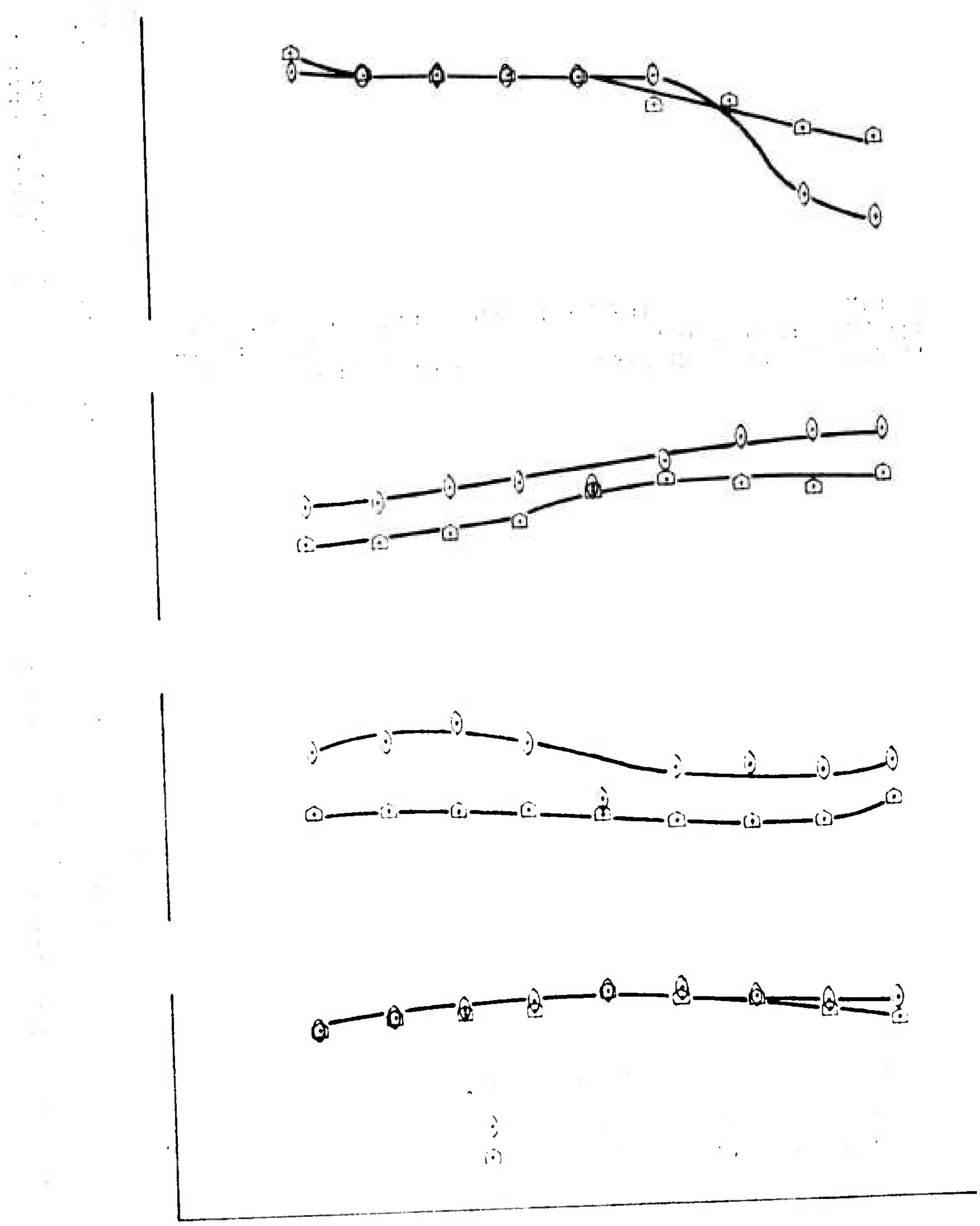


Figure D-30. Control positions for 45-degree quartering flight.

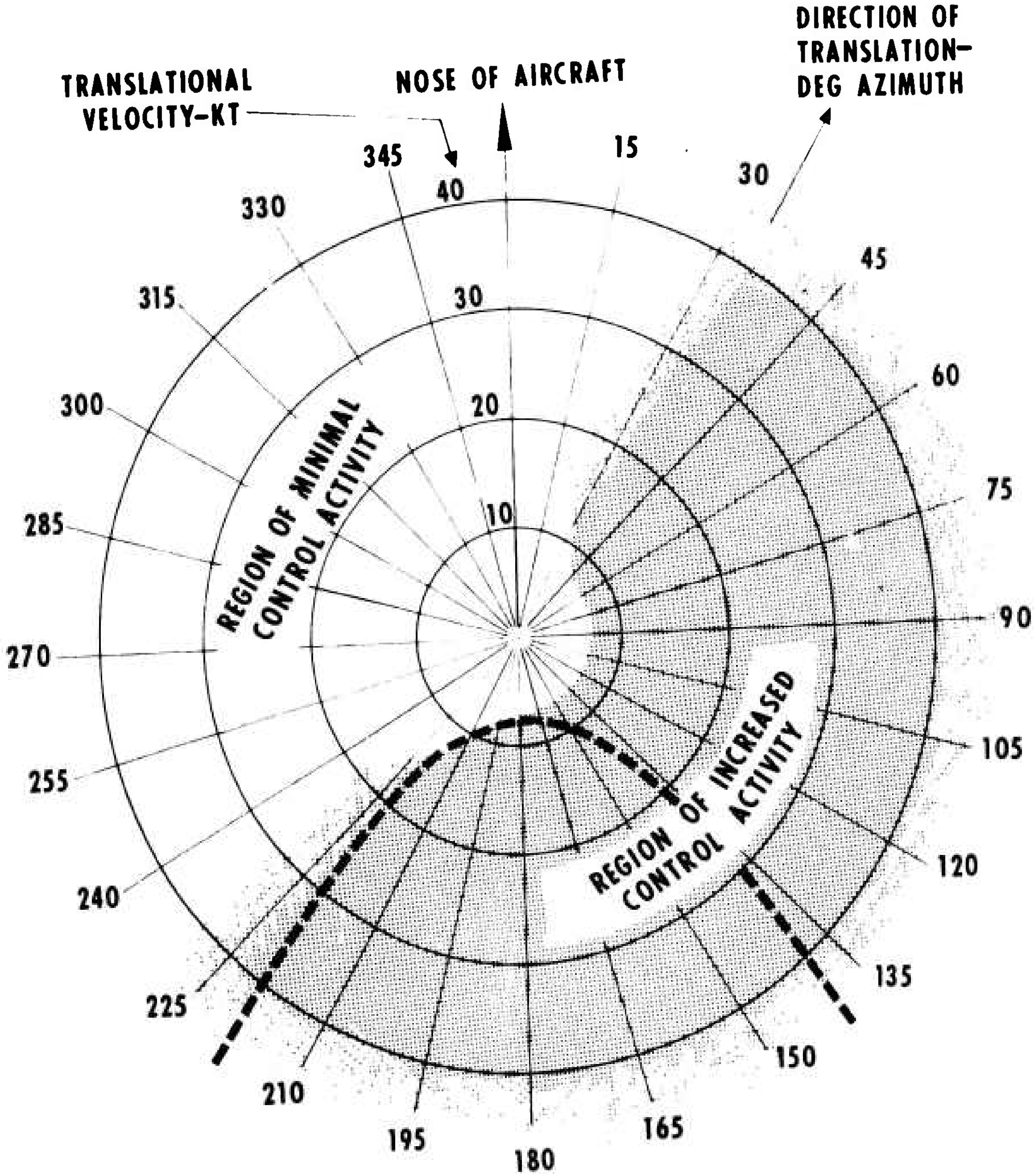


Figure D-31. Polar diagram of pilot effort required in hovering flight.

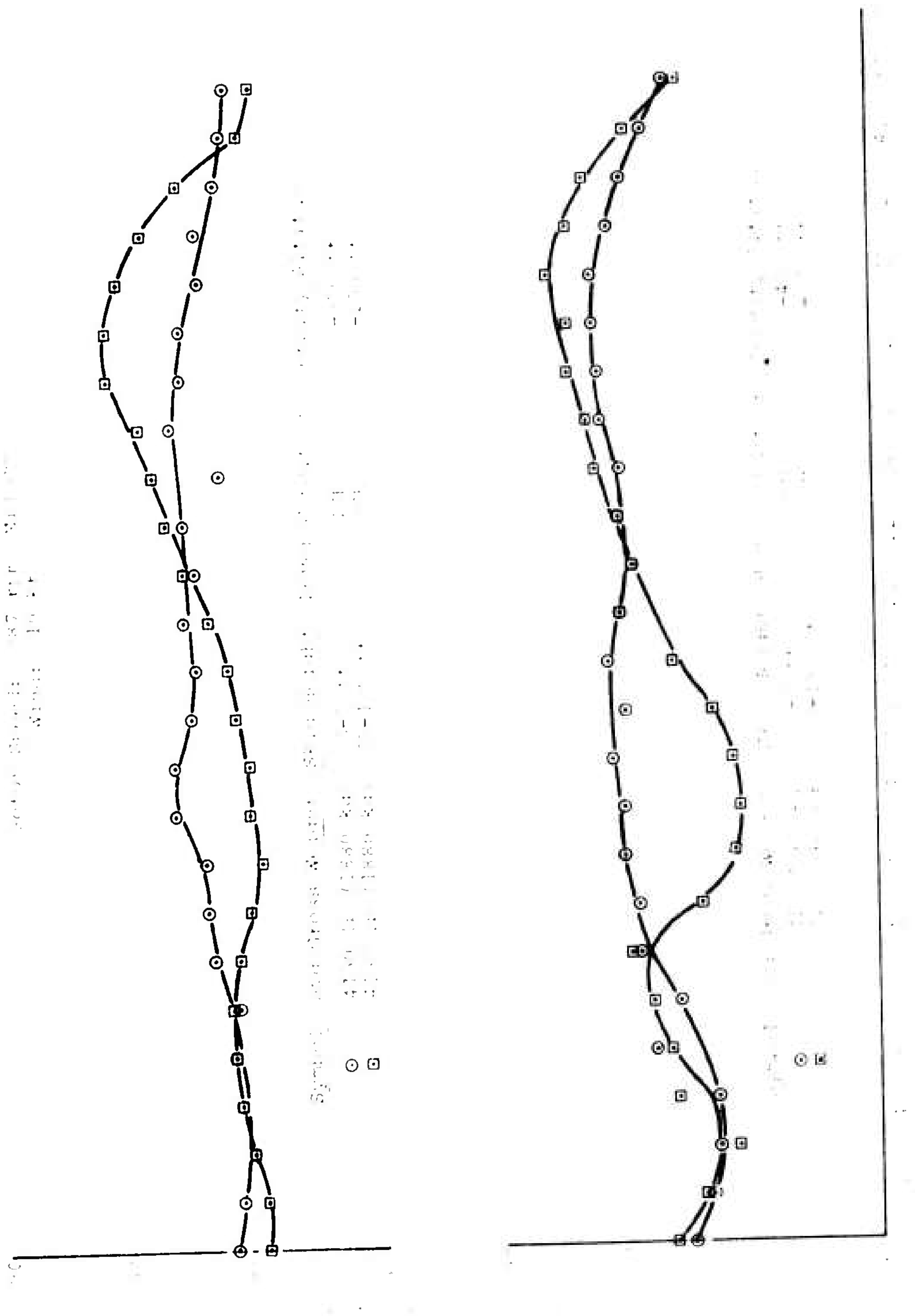


Figure D-32. Hovering pedal positions at various wind azimuths.