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# TAIL ROTOR PERFORMANCE AND TRANSLATIONAL FLIGHT HANDLING QUALITIES TESTS

## UH-1H HELICOPTER

FINAL REPORT


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JANUARY 1972




US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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## ABSTRACT

Engineering flight tests were conducted on the UH-1H helicopter to evaluate the performance and handling qualities during hover, translational flight, and forward flight. Tests were conducted in California at Edwards Air Force Base and at test sites near Bishop during the period 13 July to 9 August 1971. The UH-1H helicopter is being purchased by the US Air Force to perform search and rescue missions. Selected performance parameters and handling qualities were quantitatively and qualitatively evaluated. For the conditions tested, the UH-1H does not comply with paragraphs 3.2.1, 3.3.2, and 3.3.6 of the military specification, MIL-H-8501A. There were three deficiencies, the correction of which appears essential for adequate mission accomplishment: (1) insufficient longitudinal control within the approved gross-weight/center-of-gravity envelope, (2) insufficient directional control, and (3) directional instability between 10 and 18 knots at relative azimuths between 210 and 320 degrees.

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# INTRODUCTION

## BACKGROUND

1. The UH-1H helicopter (Air Force model designation: HH-1H) is being purchased by the US Air Force to perform search and rescue missions. Previous tests of the UH-1D/H helicopters have indicated that hover performance and in-ground-effect (IGE) translational flight capabilities of this helicopter are limited by directional control (refs 1 and 2, app A). On 12 April 1971, the US Air Force Aeronautical Systems Command (ASD) requested (ref 3) the US Army Aviation Systems Command (AVSCOM) to conduct a qualitative and limited quantitative evaluation to determine tail rotor performance and directional control margins while hovering in winds. This request was expanded on 22 June 1971 (ref 4) to include determination of: (1) main rotor hovering performance, (2) IGE handling qualities with a right lateral center-of-gravity (cg), and (3) handling qualities of the helicopter with a fire suppression kit (FSK) attached to the cargo hook. The US Army Aviation Systems Test Activity (USAASTA) was directed by AVSCOM to conduct tests to satisfy these requests (refs 5 and 6). Funding for the program was provided by ASD through a military interdepartmental purchase request (MIPR) (refs 7 and 8).

## TEST OBJECTIVES

2. Specific objectives of this test were as follows:

a. Determine main rotor and tail rotor hovering performance at 4,200 and 9,500 feet mean sea level (MSL).

b. Determine control margins at 4,200 and 9,500 feet MSL while hovering IGE in simulated wind conditions.

c. Evaluate IGE handling qualities and control margins up to 30 knots in sideward and rearward flight at 4,200 and 9,500 feet MSL with the aircraft loaded to a right lateral cg configuration.

d. Evaluate handling qualities at 4,200 feet MSL in rearward flight with the FSK installed.

e. Evaluate handling qualities and vibration characteristics during climbing, level flight, and partial power descent with the FSK installed.

## DESCRIPTION

3. The test helicopter, S/N 67-17145, is a standard production UH-1H aircraft manufactured by the Bell Helicopter Company. The main rotor is a single, two-bladed, teetering type with a stabilizer bar. A two-bladed teetering antitorque rotor is located at the top of the vertical stabilizer. The helicopter is powered by a Lycoming T53-L-13 turboshaft engine rated at 1,400 shaft horsepower (shp) at sea level (SL) under standard day, uninstalled conditions. The engine is derated to 1,100 shp due to the maximum torque limit of the helicopter's main transmission. The design gross weight of the UH-1H is 6,600 pounds, and the maximum gross weight is 9,500 pounds. A more detailed description of the UH-1H helicopter is contained in appendix B and in the operator's manual (ref 9, app A).

## SCOPE OF TEST

4. The UH-1H helicopter was evaluated to determine the hovering performance and low-speed IGE handling qualities. Handling qualities with the FSK installed were also evaluated in rearward flight IGE and forward flight at altitude. During this program, 24 flights were conducted for a total of 35.4 hours, 20.4 of which were productive test hours. All flights were performed and supported by USAASTA personnel. The testing was conducted in California from 13 July through 9 August 1971 at Edwards Air Force Base (AFB) (2,302-foot elevation) and at high-altitude test sites near Bishop (4,112- and 9,500-foot elevations). The total flight hours include ferry time between Edwards AFB and Bishop, and return, and flight time between test sites. The conditions for each test are presented in the Results and Discussion section of this report.

5. The test program was conducted within the limitations established by the AVSCOM test directives (refs 5 and 6, app A).

6. Prior to testing, the aircraft flight and engine controls were rigged in compliance with appropriate US Army publications. The swashplate was rigged to 2.0 degrees, down, left, to comply with TM 55-1520-210-20 (ref 10, app A). This swashplate rigging was chosen to compensate for the right lateral cg expected when using the rescue hoist during rescue operations. Tracking of the main rotor and tail rotor was performed prior to the start of the test program.

7. The empty weight of the test aircraft with test instrumentation installed was 5,420 pounds with the longitudinal cg at fuselage station (FS) 143.97 (aircraft battery located at FS 5.0) or FS 147.13 (aircraft battery located at FS 233.0). The lateral cg for these two conditions was buttline (BL) 0.38 left, or BL 0.36 left, respectively. The estimated weight of the test instrumentation was 63 pounds with a longitudinal fuselage station (moment arm) of 190.21.

8. The UH-1H was evaluated as a utility helicopter. Military specification MIL-H-8501A (ref 11, app A) was used to determine specification conformance.

Handling qualities ratings were assigned in accordance with the Handling Qualities Rating Scale (HQRS) presented in appendix C. Qualitative pilot comments were used to determine deficiencies and shortcomings. The applicable portion of the terms "Deficiency" and "Shortcoming," as defined in Army regulation AR 310-25 (ref 12), are presented below:

a. **Deficiency** – A defect or malfunction discovered during the life cycle of an equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued; indicates improper design or other cause of an item or part, which seriously impairs the equipment's operational capability. A deficiency normally disables or immobilizes the equipment; and if occurring during test phases, will serve as a bar to type classification action.

b. **Shortcoming** – An imperfection or malfunction occurring during the life cycle of equipment, which should be reported and which must be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the usability of the materiel or end product. If occurring during test phases, the shortcoming should be corrected if it can be done without unduly complicating the item or inducing another undesirable characteristic such as increased cost, weight.

## METHODS OF TEST

9. Test methods and data reduction procedures used in these tests are proven engineering flight test techniques and are described in appendix D. Tests were conducted in nonturbulent atmospheric conditions, unless otherwise stated, so the data would not be influenced by uncontrolled disturbances. The flight test data were manually recorded from test instrumentation located in the copilot panel and engineer auxiliary test panel. A list of the test instrumentation is included as appendix E.

## CHRONOLOGY

10. The chronology of this test program is as follows:

| | | | |
|---|---|---|---|
| Test directive received | 11 | June | 1971 |
| Safety-of-flight release received | 11 | June | 1971 |
| Test instrumentation installation begun | 21 | June | 1971 |
| Flight test started | 13 | July | 1971 |
| Flight test completed | 9 | August | 1971 |

## RESULTS AND DISCUSSION

### GENERAL

11. Engineering flight tests were conducted on the UH-1H helicopter to evaluate the performance and handling qualities during hover, translational flight, and forward flight. Selected performance parameters and handling qualities were quantitatively and qualitatively evaluated. For the conditions tested, the UH-1H did not comply with paragraphs 3.2.1, 3.3.2, and 3.3.6 of MIL-H-8501A (ref 11, app A). There were three deficiencies, the correction of which appears essential for adequate mission accomplishment: (1) insufficient longitudinal control within the approved gross-weight/cg envelope, (2) insufficient directional control, and (3) directional instability between 10 and 18 knots at relative azimuths between 210 and 320 degrees.

### PERFORMANCE

#### Antitorque System

12. Antitorque system performance tests were conducted to determine the limitations of aircraft performance which were attributable to the antitorque system. An instrumented 90-degree tail rotor gearbox was utilized to measure tail rotor torque. All antitorque system performance test data were acquired in conjunction with other tests. Results of these tests are presented in figures 1 through 15, appendix F.

13. A directional control margin of 10 percent of the full displacement was qualitatively determined to be the minimum acceptable to provide adequate control of the helicopter. No controllability data are available on the UH-1H helicopter to determine the magnitude of yaw response associated with this control margin. Additional testing would be required to determine the controllability characteristics of the UH-1H helicopter.

14. Figure 1, appendix F, presents the variation in directional control margin as a function of skid height and main rotor thrust coefficient ($C_T$). The skid height at which minimum $C_T$ (hence minimum gross weight) occurred varied between out of ground effect (OGE) and 10 feet, depending on the magnitude of the directional control margin. The $C_T$ associated with a directional control margin of 30 percent increased steadily with decreasing skid height. Nonlinear relationships between skid height and $C_T$ were noted for directional control margins less than 25 percent.

15. The tail rotor blades were rigged to yield an average blade pitch angle of 18 degrees with full left pedal. A plot of average tail rotor blade angle versus directional control position is presented in figure 15, appendix F. The average tail rotor blade pitch angle is essentially linear as a function of pedal position.

16. The power required at the output shaft of the 90-degree tail rotor gearbox for various directional control margins at different skid heights is shown in figures 9 through 11, appendix F. The tail rotor power required did not vary significantly as a function of skid height for directional control margins of 10 percent or more. The maximum difference in tail rotor power was less than 20 horsepower between a 2-foot hover and OGE with a 10-percent directional control margin at SL, standard-day conditions. However, at directional control margins of less than 10 percent, the tail rotor power required varied nonlinearly. It was calculated that tail rotor powers in excess of 170 horsepower should be anticipated during hover at directional control margins of 5 percent or less while operating at or near SL. The tail rotor power for a given blade pitch angle varies as a function of density altitude, and decreased as density altitude increased. The structural design criteria report (ref 13, app A) for the UH-1H states that the antitorque drive system design limit is 386 foot-pounds (ft-lb) of torque (122 shp at 1,654 rpm).

17. The percentage of total engine power that was required by the tail rotor as a function of directional control position during a hover is presented in figure A. This percentage varied as a function of skid height and directional control position. The skid heights requiring the largest and smallest percentages were 10 feet and OGE, respectively. Increasing directional control between 35 and 10 percent caused the antitorque drive system power percentage to increase almost linearly. The percentage of total engine power absorbed by the tail rotor varied nonlinearly when the directional control margin was less than 10 percent. This nonlinear increase in power distribution to the tail rotor indicated the presence of tail rotor blade stall. An increase in aircraft high-frequency vibration was noted when a left directional control margin of less than 10 percent was encountered. This increase in aircraft high-frequency vibration also indicated that some form of tail rotor blade stall was being encountered. The severity of the high-frequency vibration increased as the directional control approached the left control limit.

## Hovering

18. The objective of these tests was to determine hovering performance as a function of skid height by using the tethered hover method. During these tests, the longitudinal cg varied from FS 136.2 to FS 138.0. The test results are presented in figures 16 through 23, appendix F. The faired lines on these figures were derived from the YUH-1D performance report (ref 2, app A). The hovering test conditions are presented in table 1.

19. The skid height for each test was determined by measuring the distance from the bottom of the left landing gear skid tube to the ground. The reference point for all skid height measurements was BL 48.0 left, water line (WL) -7.0, and FS 134.5, which is opposite the cargo hook (photo 1). All hovering tests were conducted with the engine inlet screens and particle separator installed. This inlet configuration was necessary to prevent foreign object damage to the engine which could have occurred at the Bishop, California, test sites.

NOTES: 1. Dashed portion of curves indicates extrapolated data.
2. Density altitude = 10,000 feet.
3. Rotor speed = 324 rpm.

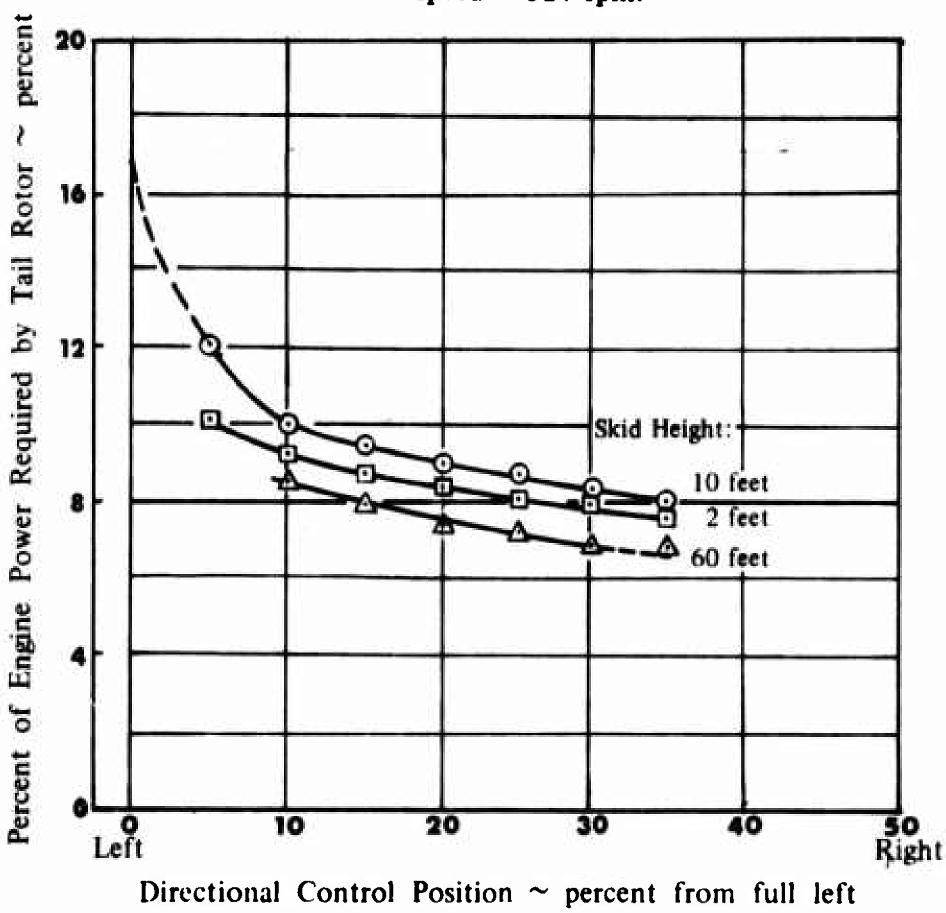



Figure A. Tail Rotor Power Fraction.

Table 1. Hovering Performance Test Conditions.

| Skid Height (ft) | Terrain Altitude Above Mean Sea Level (ft) | Rotor Speed (rpm) |
|---|---|---|
| IGE: 2, 5, 10, 15, 20, and 30<br>OGE: 60 | 4,220 and 9,550 | 324 and 314 |

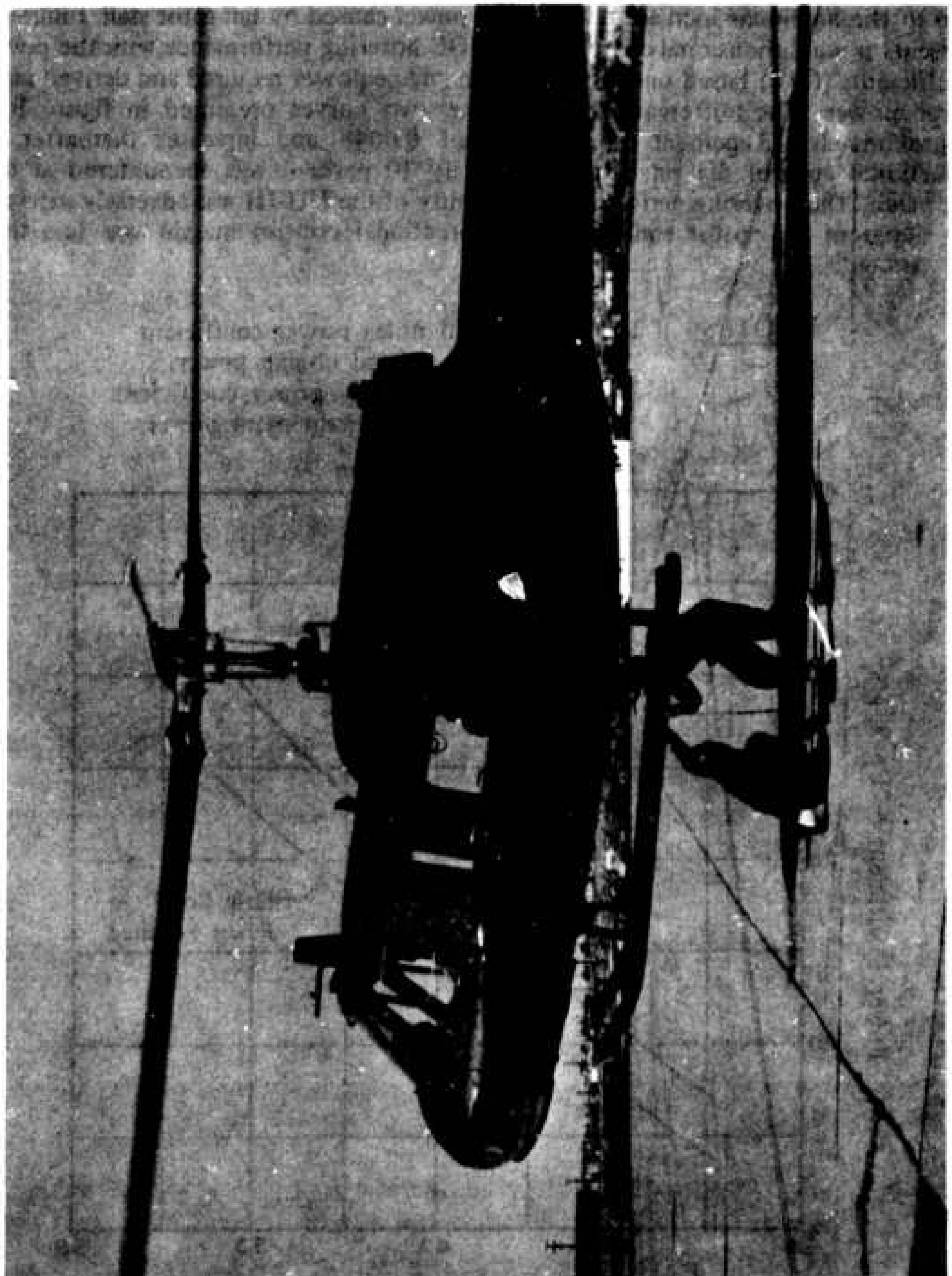


Photo 1.   Determination of Hovering Skid Height.

20. The percentage of total engine power available to the main rotor decreased due to the nonlinear increase in tail rotor power caused by tail rotor stall. Figure B presents a nondimensional comparison of IGE hovering performance with the power coefficients (Cp's) based on both measured engine power required and derived main rotor power. The difference between the two curves presented in figure B is approximately 13 percent up to a $C_T$ of 0.0048 and increases thereafter. A directional control margin of approximately 10 percent was encountered at this $C_T$ value. The hovering performance capability of the UH-1H was adversely affected by apparent tail rotor stall when the directional control margin was less than 10 percent.

NOTES:
1. Solid line denotes power coefficient based on measured engine power.
2. Dashed line denotes power coefficient based on derived main rotor power.

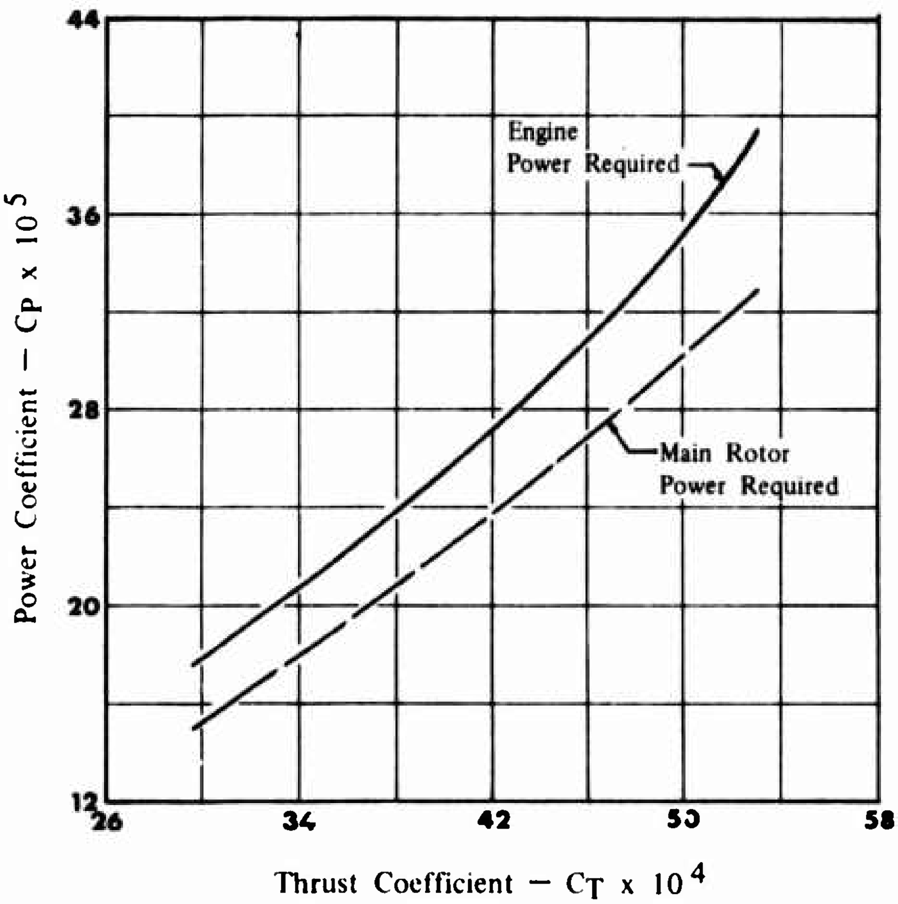



Figure B. Hovering Performance at a 2-Foot Skid Height.

21. The IGE hovering performance is limited by directional control in many areas depending on skid height, gross weight, density altitude, and rotor speed. The most critical skid height for a directional control margin of 10 percent is 10 feet. Figure C presents the IGE, standard-day hovering capability of the UH-1H helicopter at a skid height of 10 feet and a rotor speed of 324 rpm. The hovering capability is reduced at altitudes above 9,000 feet when observing the recommended 10-percent directional control margin.

NOTES:
1. Standard day.
2. Wind less than 2 knots.
3. Skid height = 10 feet.
4. Rotor speed = 324 rpm.
5. Engine particle separator installed.

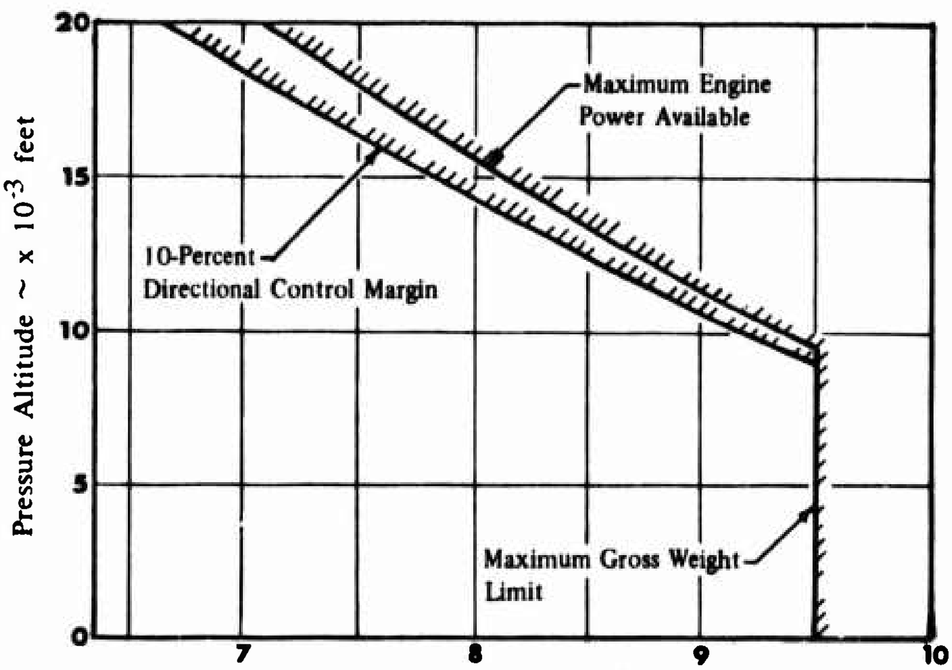



Figure C. IGE Hovering Performance.

22. The IGE hovering performance is further degraded when hovering in adverse crosswind conditions. The limitations and handling qualities during translational flight are discussed in paragraphs 27 through 32.

## Level Flight Performance With Fire Suppression Kit

23. The objective of this test was to define the approximate increase in equivalent flat plate area with the FSK installed (photo 2). The results of this test are

presented in figure 24, appendix F. It should be noted that the atmospheric conditions during this test were unacceptable, and additional testing would have to be conducted to verify the calculated increase in equivalent flat plate area. It was calculated from the frontal area and a drag coefficient of 1.0 that the equivalent flat plate area increased approximately 6 square feet with the FSK installed. The level-flight performance data, presented in reference 1, appendix A, were used as the baseline against which to determine this increase in equivalent flat plate area.

## HANDLING QUALITIES

### Cyclic Pitch Control Pattern

24. The cyclic pitch control pattern was determined during ground tests with the rotor in a static position. Hydraulic and electrical power were provided by external sources. The test was conducted with the collective control in the full-down position. The center of the cyclic control grip was used as a reference in determining the magnitude of the cyclic control displacement. A plot of the cyclic pitch control pattern is presented in figure 25, appendix F. The cyclic pitch control pattern shows that the available longitudinal and lateral control are mutually dependent. This mutual dependence usually occurs when either the longitudinal or lateral control is within 15 to 25 percent of the control limit. This plot should be used when determining longitudinal and lateral cyclic control margins.

### Translational Flight Evaluation

25. Translational flight is defined as flight in any direction with relative wind at any azimuth from zero through 360 degrees (measured clockwise from nose of aircraft) at airspeeds between zero and 35 knots. The primary objectives of this test were to evaluate the handling qualities and to determine control margins in translational flight. The applicable subparagraphs of paragraphs 3.2 and 3.3 of MIL-H-8501A (ref 11, app A) were used as a basis for evaluation. A secondary purpose of this test was to determine the tail rotor power required to stabilize the aircraft at various combinations of wind azimuth and wind speed. The test method used to meet these objectives was to conduct translational flights at various combinations of relative azimuth and speed by using a calibrated ground pace vehicle as a reference. When the aircraft was stabilized in translational flight, parameters necessary to determine gross weight, ambient air conditions, azimuth, ground speed, and control margins were recorded. The wind velocity, measured approximately 10 feet above the ground, was less than 4 knots during all of the tests. Ambient wind speed and direction were incorporated into the analysis when determining the vectorial airspeed summation and relative azimuth to the nose of the aircraft. A nearly constant main rotor thrust coefficient was maintained for each test condition by adding ballast as fuel was consumed. The test conditions evaluated are listed in table 2. Results of the translational flight handling qualities are graphically presented in figures 26 through 66, appendix F.

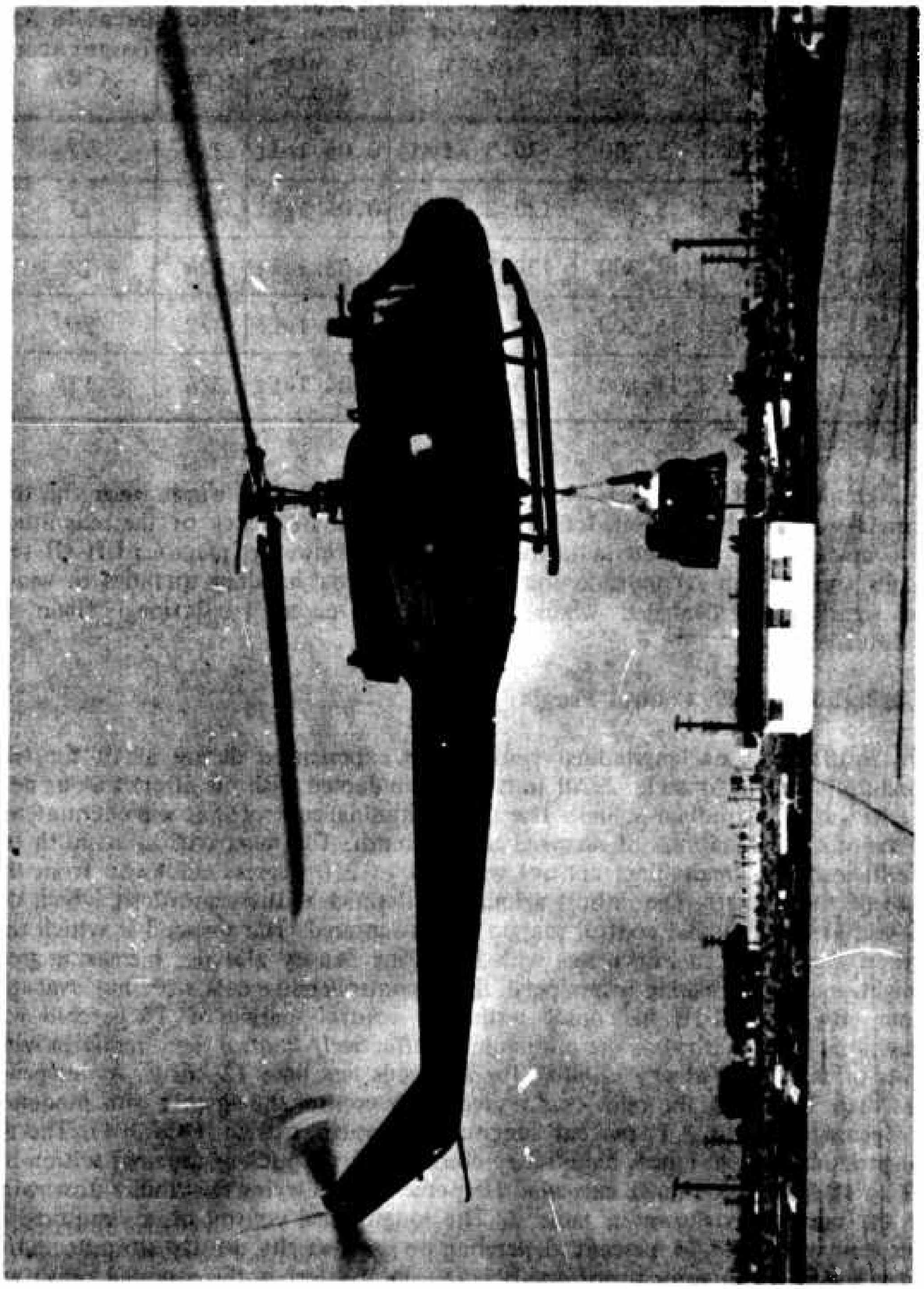


Photo 2. UH-1H with Fire Suppression Kit (FSK) Attached to Cargo Hook.

Table 2. Translational Flight Handling Qualities Test Conditions.

| Test Condition | Gross Weight (lb) | Density Altitude (ft) | Longitudinal Center of Gravity (in.) | Lateral Center of Gravity (in.) | Rotor Speed (rpm) | Outside Air Temperature (°C) |
|---|---|---|---|---|---|---|
| 1 | 7,400 | 5,200 | 130.5 (fwd) | 0.05 left | 323 | 17 |
| 2 | 8,600 | 5,100 | 130.2 (fwd) | 0.04 left | 322 | 17 |
| 3 | 7,600 | 11,300 | 130.3 (fwd) | 0.05 left | 324 | 12 |
| 4 | 9,400 | 5,600 | 134.0 (fwd) | 0.04 left | 323 | 20 |
| 5 | 8,600 | 11,500 | 130.5 (fwd) | 0.04 left | 324 | 13 |

26. Translational flight tests were not conducted in actual winds since the test aircraft was not instrumented to record either the frequency or the magnitude of control inputs required in unsteady air flow. However, previous UH-1H test results (ref 1, app A) indicate that the translational handling qualities in winds are significantly degraded when compared to paced translational flight in undisturbed air.

**Longitudinal Cyclic Control Margin:**

27. Insufficient aft longitudinal control was experienced during all of the test conditions shown in table 2. All tests were conducted with the aircraft at or near the forward longitudinal cg limit. The aft longitudinal control limit was encountered at various combinations of airspeed and azimuth. The most critical azimuth for insufficient aft longitudinal control was 190 to 210 degrees clockwise from the nose of the aircraft. The critical azimuth is defined as the azimuth at which the 10-percent longitudinal control margin is encountered. The airspeed at which this critical azimuth occurs decreases with increasing density altitude, increasing gross weight, and/or decreasing rotor speed. Since controllability data were not available from previous UH-1H helicopter testing, a control margin of 10 percent was qualitatively established as the minimum to effectively control the aircraft in wind gusts of ±2 knots at any azimuth for airspeeds less than 12 knots. At airspeeds less than 12 knots, the pilot could adequately control the aircraft with moderate compensation with a 10-percent longitudinal control margin (HQRS 4). The aft longitudinal control inputs required to control aircraft pitching motions within the 12 to 18-knot speed range exceeded 10 percent when trying to stabilize downwind at the conditions shown in table 2. The longitudinal control inputs required in this area were 8 to 15 percent, depending on gross weight, density altitude, and/or rotor speed (main rotor thrust coefficients). Pilot effort in this airspeed range was excessive when trying to achieve adequate performance (HQRS 6). Prolonged

exposure (5 to 10 minutes) at these flight conditions induced pilot fatigue and decreased pilot efficiency. The pilot effort associated with prolonged exposure was intense, and adequate performance is unattainable (HQRS 7). As the speed of the aircraft increased from 18 to 30 knots, the 10-percent aft longitudinal control margin was sufficient to control the helicopter in translational flight. Minimal pilot compensation was required to obtain desired performance at these higher speeds (HQRS 3).

28. The aft longitudinal control capability was significantly less than that required to meet the intent of paragraph 3.2.1 of MIL-H-8501A. This insufficient aft longitudinal control problem is only encountered when the aircraft is operating at or near the forward longitudinal cg limit. It was recommended in reference 1, appendix A, that a precautionary loading envelope (fig. D) be incorporated in the operator's manual. That proposed envelope was not investigated during this test program. However, based on the results of this test program, a change in the cg envelope is needed. Without an appropriate cg envelope change, the lack of sufficient aft longitudinal control within the present gross-weight/cg envelope is a safety-of-flight hazard, and improvement is mandatory to meet the intended mission.

**Directional Control Margin:**

29. The directional control capability was signficantly less than that required to meet the intent of paragraphs 3.3.2 and 3.3.6 of MIL-H-8501A. The critical area of insufficient directional control (less than 10-percent control margin) for test condition number 1, table 2, was bounded by azimuths from approximately 30 to 45 degrees at an airspeed of 23 knots with a skid height of approximately 15 feet. The area of inadequate directional control bounded by azimuth and speed increased with combinations of increasing altitude, increasing gross weight, and/or decreasing rotor speed. At the most critical condition (number 5, table 2), the left directional control margin was less than 10 percent at a wind speed of less than 10 knots for azimuths between 75 and 265 degrees. For this same condition, the directional control margin to the right was less than 10 percent at wind speeds greater than 25 knots from the left. A directional control margin of at least 10 percent allowed the pilot, with considerable compensation, to stabilize or maneuver the aircraft in translational flight (HQRS 5). Increased directional control is mandatory if mission accomplishment is to be achieved within the currently approved flight envelope of the aircraft.

30. The UH-1H was difficult to stabilize directionally at speeds between 10 and 18 knots at relative azimuths between 210 and 320 degrees. Figure E illustrates a representative area of this directional instability. Rapid, and sometimes large, directional control excursions were necessary to maintain a heading for these conditions. Pilot recognition and reaction times following small excursions in yaw determined the frequency and magnitude of the directional control inputs. During test condition number 3, table 2, directional control of the aircraft was lost (uncontrolled right yaw) when trying to stabilize at a speed of 13 knots at a relative azimuth of 270 degrees (left crosswind) (HQRS 10). Available directional control

was insufficient to arrest the yawing motion until the helicopter had yawed uncontrollably 300 degrees to the right. Insufficient directional control precludes safe operation within the currently approved flight envelope, and improvement in directional control capability is mandatory.

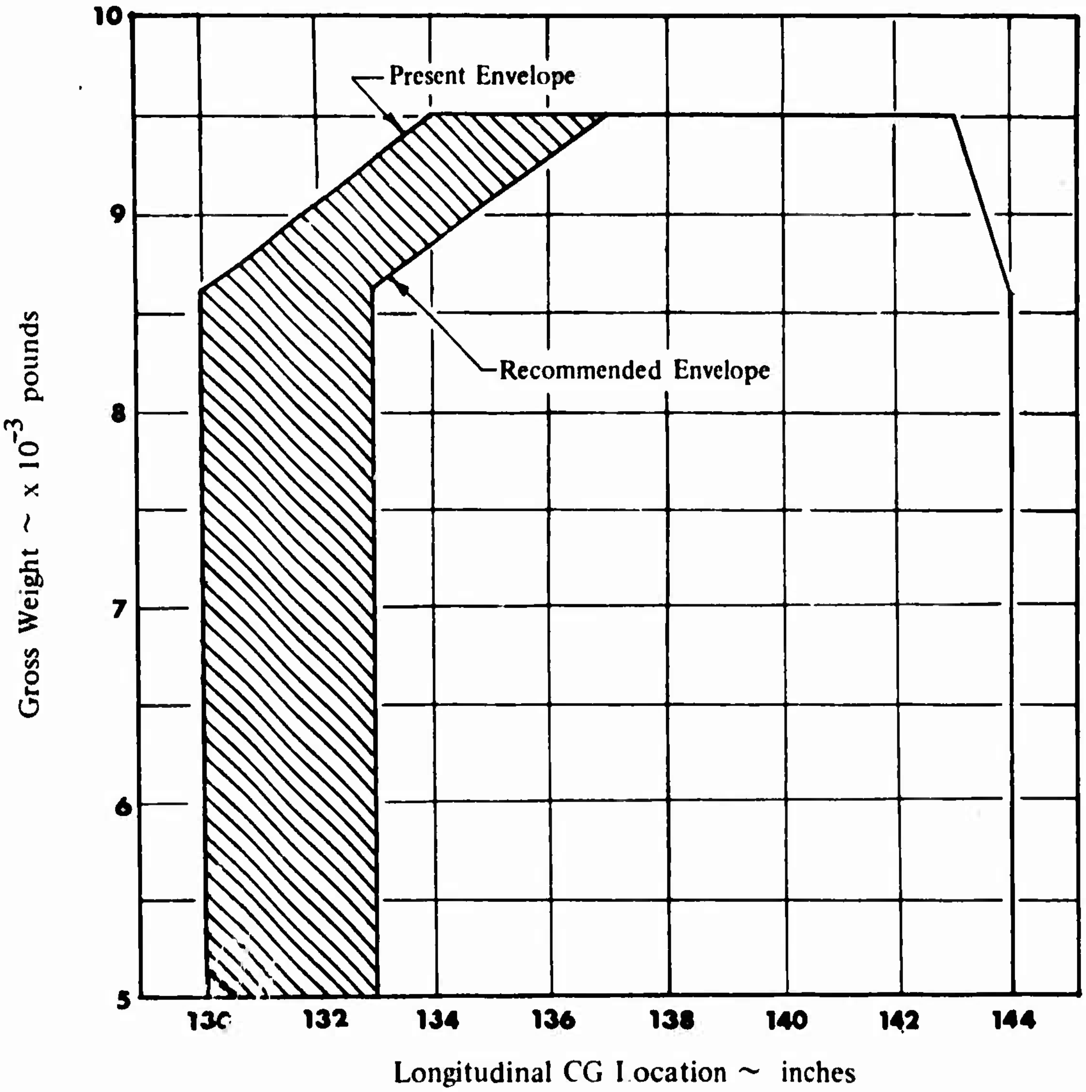



Figure D. Recommended Gross-Weight/CG Envelope.

| Legend | Speed (kt) | Density Altitude (ft) | Gross Weight (lb) | Coefficient of Thrust |
|---|---|---|---|---|
| ——— | 11.0 | 5,180 | 7,430 | 0.003057 |
| – – – – | 19.5 | 5,150 | 7,220 | 0.002970 |

NOTES:
1. Aircraft could not be stabilized between 120 and 210 degrees due to insufficient longitudinal control at 19.5 knots.
2. Longitudinal center of gravity = FS 130.6 (fwd).
3. Lateral center of gravity = 0.05 left.
4. Rotor speed = 324 rpm.
5. Skid height range = 5 to 15 feet.

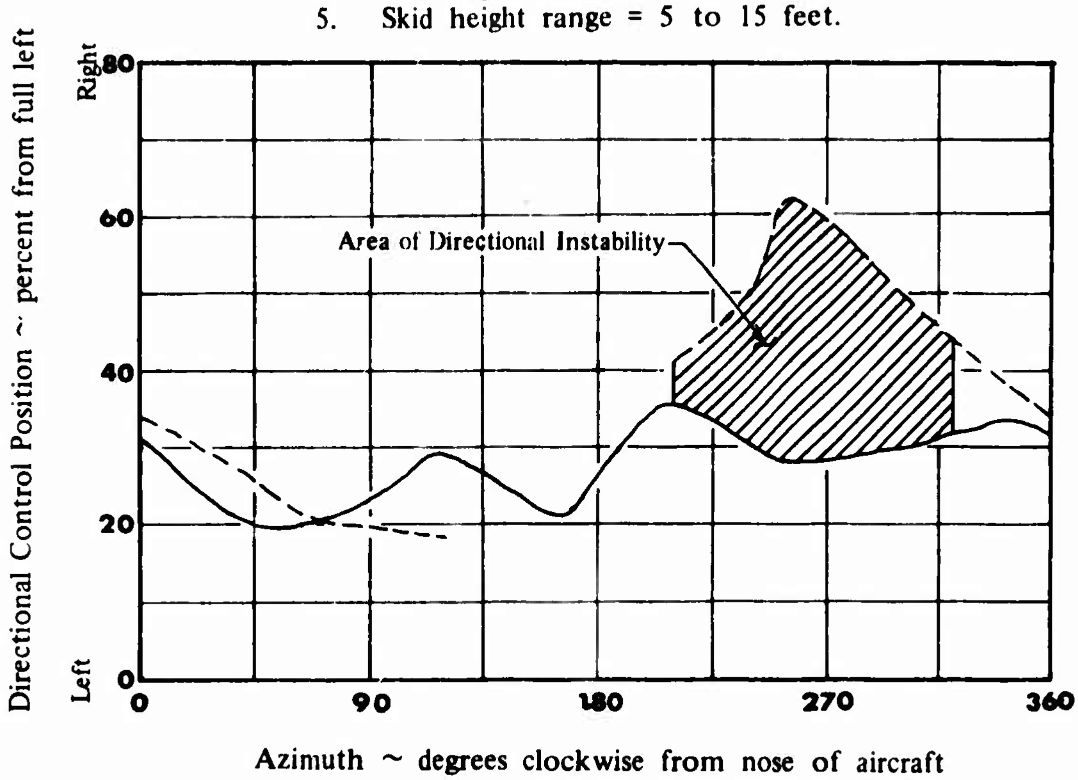



Figure E. Directional Control Position IGE at Various Wind Azimuths.

31. The lack of sufficient directional control can be further complicated by the large changes in longitudinal cyclic control required to arrest the pitching motions of the aircraft. Following the loss of directional control discussed in paragraph 30, the helicopter started to pitch nose down as it rotated through a heading of 240 degrees. Full aft cyclic control was required to check the pitching motions as the aircraft continued to turn to the right. Insufficient directional control, when combined with insufficient longitudinal control, is a safety-of-flight hazard, and correction or imposition of appropriate limitation (fig. F) is mandatory for safe operation.

NOTES:
1. Rotor speed = 324 rpm.
2. Envelope based on a 10-percent control margin at all wind azimuths.
3. Longitudinal center of gravity at or near forward limit.
4. Curves derived from figure 26, appendix E.

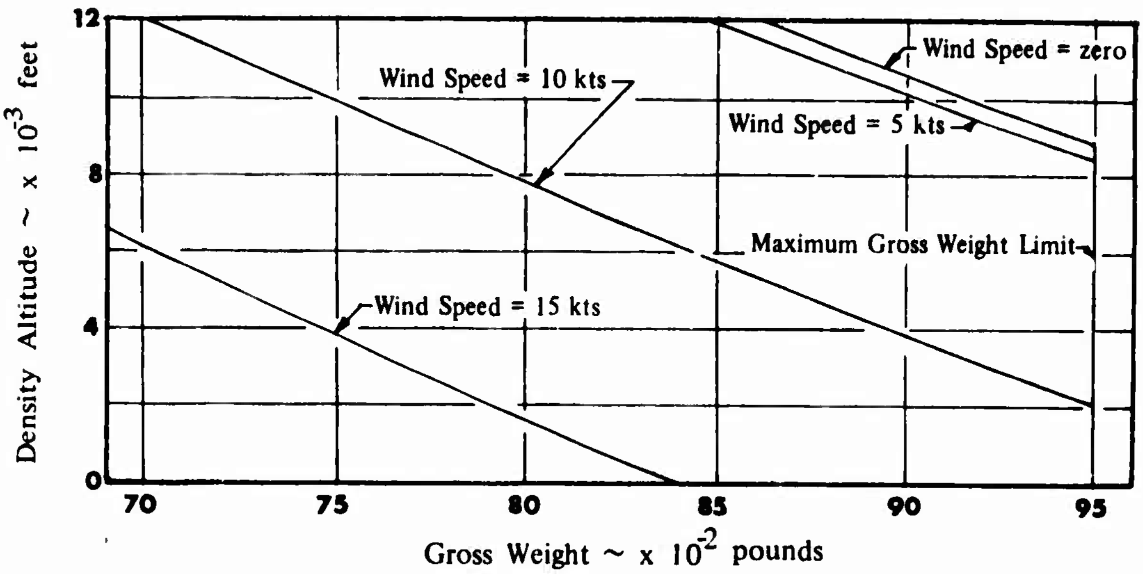



Figure F. Recommended IGE Translational Flight Envelope for the Present Gross-Weight/CG Flight Envelope.

32. The IGE translational flight envelope for 10-percent longitudinal and directional control margins is presented in figure F for the present gross-weight/cg envelope. When the aircraft is loaded to provide a 10-percent longitudinal control margin in translational flight, the helicopter is still limited by a 10-percent directional control margin, as shown in figure G. It is required that the directional control system for the UH-1H be improved to provide increased translational flight capabilities.

**Lateral-Directional Characteristics:**

33. No undesirable lateral control characteristics were encountered during the translational flight handling qualities evaluation. There was adequate lateral control margin to control the aircraft in roll with a mid lateral cg loading. However, an undamped lateral-directional oscillation with a period of about 1.5 to 2.0 seconds was encountered at airspeeds greater than 25 knots in right translational flight IGE at thrust coefficients greater than 0.0035. This lateral-directional oscillation was only encountered in those translational flight regimes where full left pedal

was required. The helicopter oscillated approximately ±10 degrees in yaw and approximately ±2 degrees in roll. The oscillations were unpleasant, since a desired heading could not be precisely maintained (HQRS 3). An attempt was made to excite this motion OGE, but all attempts failed to induce any lateral-directional oscillation at this condition.

**Aircraft Height Control:**

34. Control of skid height by use of collective control was easilv accomplished (HQRS 2).

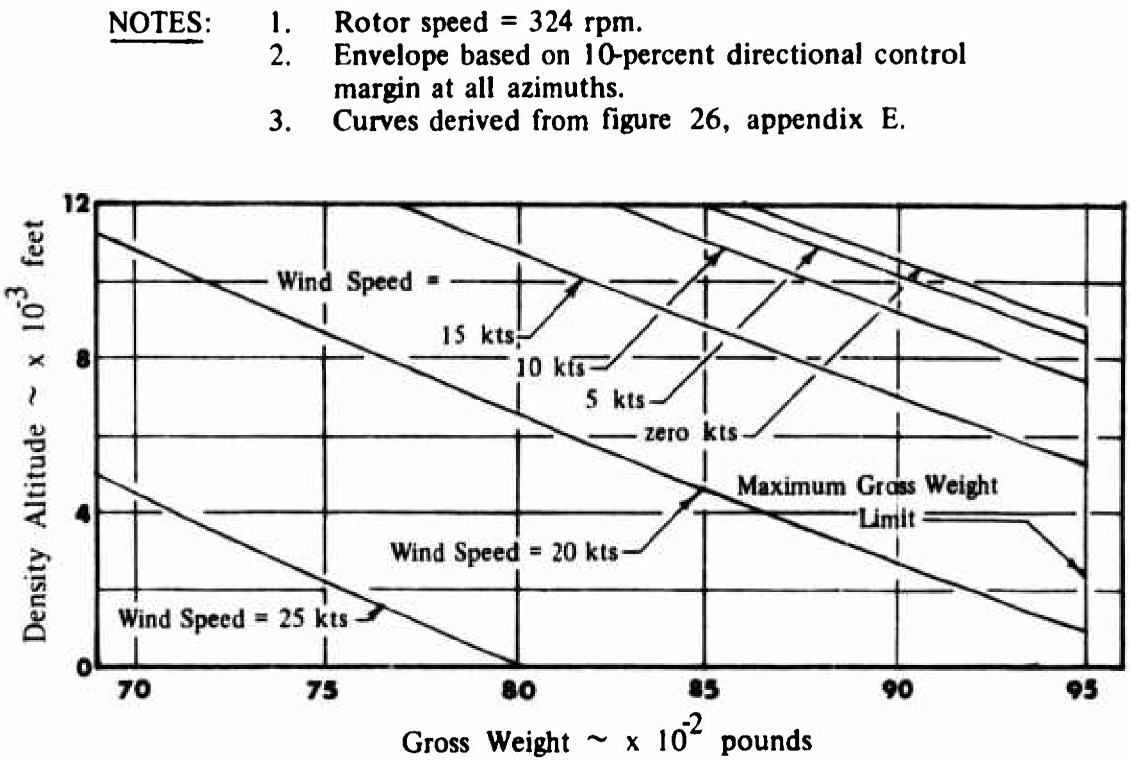



Figure G. Recommended IGE Translational Flight Envelope for Reduced Forward CG Flight Envelope.

**Tail Rotor Power:**

35. The tail rotor power required to stabilize the aircraft increased nonlinearly as the directional control approached the left limit. The maximum tail rotor power encountered during translational flight was 140 horsepower for test condition number 2 and number 4, table 3. The rapid, and sometimes large, directional control excursions discussed in paragraph 30 caused large tail rotor power oscillations. Oscillating tail rotor power levels were not measured exactly during this program but are estimated to vary from 40 to 140 horsepower for some test

conditions. At no time during the test program was any of the tail rotor drive train (gearboxes, shafting, etc.) replaced. It has been noted on previous test programs that higher tail rotor horsepower was recorded during translational flight or controllability tests at sea level with the aircraft loaded at maximum gross weight (ref 14, app A). Therefore, additional testing is required to determine tail rotor drive system power required at or near sea level.

## Right Lateral Center-of-Gravity Evaluation

36. The primary objectives of this test were to evaluate the handling qualities and to determine control margin with a right lateral cg loading. A secondary objective of this test was to determine the tail rotor power required to stabilize the aircraft in sideward flight when loaded asymmetrically. Various lateral asymmetric loadings were used to simulate rescue missions with the hoist located on the right side at longitudinal FS 80.0. All tests were conducted in sideward and rearward flight within a 5- to 15-foot skid height range. A calibrated ground pace vehicle was used as a speed reference. Conditions tested are listed in table 3. Results of these tests are presented in figures 67 through 76, appendix F.

Table 3. Lateral Center-of-Gravity Test Conditions.

| Gross Weight (lb) | Density Altitude (ft) | Rotor Speed (rpm) | Longitudinal Center of Gravity (in.) | Lateral Center of Gravity (in.) |
|---|---|---|---|---|
| 7,600 | 6,000 | 323.5 | 130.5 (fwd) | 0.05 left to 4.38 right |
| 8,400 | 5,300 | 323.0 | 130.5 (fwd) | 0.04 left to 4.01 right |
| 7,600 | 11,600 | 324.0 | 130.3 (fwd) | 0.05 left to 2.89 right |

### Lateral Cyclic Control Margin:

37. Lateral cyclic control variation with speed in sideward flight was stable (increasing right lateral cyclic control was required with increasing speed in right sideward flight) for all conditions tested. Approximately 4.5 percent (0.6 inch) of lateral cyclic control displacement was required for each 20-knot change in lateral speed. It was determined from figure 67, appendix F, that approximately 8 percent (1 inch) of lateral cyclic control displacement was required to balance a 14,000 in.-lb change in lateral moment. There was sufficient lateral cyclic control available to maneuver the aircraft during sideward flight for the conditions tested. Pilot compensation to control rolling motions was not a factor to achieve desired performance.

38. A left lateral control margin of 18.5 percent was encountered when hovering (zero wind speed) the aircraft with a right lateral moment of 35,000 in.-lb. Approximately 6.9 percent of additional left lateral cyclic displacement was required in left sideward flight when the speed was increased from zero to 30 knots. Therefore, a left lateral cyclic control margin of approximately 11.6 percent (18.5 percent minus 6.9 percent) is available in left sideward flight at 30 knots with a right asymmetric moment of 35,000 in.-lb. This calculated lateral control margin is slightly greater than the intent of paragraph 3.3.4 of MIL-H-8501A and should allow for variations in rigging between aircraft. The UH-1H should be limited to a right lateral moment of 35,000 in.-lb to provide at least a 10-percent lateral control margin. Additional tests would be required to determine the maximum allowable right lateral moment when the rigging of the main rotor swashplate is different than 2.0 degrees down, left.

39. No definite trend of lateral cyclic control requirements in rearward flight could be determined from the qualitative test results. In rearward flight, the lateral cyclic control sometimes moved to the right nonlinearly as speed was increased, while in some instances no change in lateral cyclic control was observed. Lateral control characteristics and lateral control margins in rearward flight were acceptable at all conditions tested.

**Longitudinal Cyclic Control Margin:**

40. Increasing aft longitudinal control displacement was required to stabilize in left sideward flight as speed was increased from zero to the maximum tested. The gradient of aft longitudinal control displacement was approximately 0.7 percent per knot (0.08 in./kt) in left sideward flight. A longitudinal control margin of less than 10 percent was encountered several times during the test program at speeds in excess of 20 knots in left sideward flight. However, if the proposed gross-weight/longitudinal cg envelope discussed in paragraph 29 is observed, sufficient longitudinal control should be available to control the helicopter and provide a 10-percent aft longitudinal control margin in left sideward flight. No appreciable change in longitudinal cyclic was required to maintain the trim pitch attitude when translating to the right from a hover to a speed of 10 to 15 knots. As speed exceeded 10 to 15 knots in right sideward flight, an increase in aft longitudinal cyclic was necessary to control pitch attitude for most conditions tested. The longitudinal control margin during right sideward flight was generally 15 to [illegible] percent at the limit speed. Variations in lateral cg, density altitude, and gross weight did not significantly affect the longitudinal control gradient in either left or right sideward flight. The longitudinal cyclic control characteristics were not objectionable during sideward flight and control of the aircraft could be maintained with minimal pilot compensation (HQRS 3).

41. A nonlinear increase in aft longitudinal control displacement was required to stabilize the helicopter as speed was increased in rearward flight. Variations in right lateral cg had negligible effects on the longitudinal control displacement characteristics in rearward flight. The maximum ground speed attainable in rearward flight at a 10-percent longitudinal control margin varied from 9 to 18 knots,

depending on density altitude, gross weight, and longitudinal cg. The pilot compensation required to control the aircraft ranged from moderate to intense, depending on the speed in rearward flight and time exposed to a specific flight speed. Paragraph 27 discusses in detail the longitudinal control problems encountered during rearward flight. Adoption of the recommended gross-weight/longitudinal-cg envelope (para 28) will increase the aft longitudinal control margin and the downwind hovering capabilities of the UH-1H.

**Directional Control Margin:**

42. There was a nonlinear variation in directional control position as the speed of the aircraft was increased in sideward flight. Increasing left directional control displacement was required as speed increased from zero to 10 knots in left sideward flight. This increase in left directional control with increasing left sideward speed indicated that the aircraft had a tendency to turn downwind. This flight characteristic was not objectionable up to speeds of 10 knots, and control of the helicopter could be maintained with minimal pilot effort (HQRS 3). At a speed of 10 knots in left sideward flight, a reversal in directional control requirement occurred, since increasing right directional control displacement was required to stabilize the aircraft in left sideward flight. The maximum directional control gradient in left sideward flight occurred at a speed of approximately 15 knots and had an average magnitude of 6.5 percent per knot (0.45 in./kt). Intense pilot compensation was required to stabilize the aircraft between 10 and 18 knots on a precise heading in left sideward flight (HQRS 8). At airspeeds in excess of about 18 knots, the pilot compensation was reduced to a minimal level (HQRS 3). Generally, increasing left directional control was required with increasing speed in right sideward flight. For many conditions tested, the recommended 10 percent of directional control margin (para 29) was encountered prior to reaching 30 knots in sideward flight to the right. Pilot effort and handling qualities ratings during right sideward flight were the same as those presented in paragraph 29. The variation in right lateral cg had a negligible effect on directional control requirements in sideward flight. The recommended IGE translational flight envelope presented in paragraph 32 is valid when operating at right asymmetric moments up to 35,000 in.-lb.

43. Variations in directional control as a function of speed in rearward flight exhibited no definite trend. For most conditions tested, increasing right directional control was required as rearward speed was increased. However, for several isolated test conditions, increasing left pedal was required to stabilize the aircraft in rearward flight as speed increased. Pilot compensation required to control the helicopter, along with associated handling qualities ratings during rearward flight, are the same as paragraphs 30 and 31. Again, as during sideward flight, variations in lateral cg had a negligible effect on the directional handling qualities during rearward flight.

**Tail Rotor Power:**

44. Increasing tail rotor power was required with increasing left directional control displacement both in rearward and sideward flight. The average magnitude of these

peak values was 113 and 102 horsepower at density altitudes of approximately 6,000 and 11,000 feet, respectively, with a 10-percent directional control margin. Variations in lateral cg had a negligible effect on a tail rotor power required as a function of directional control position when the directional control margin was more than 10 percent.

**Evaluation with Fire Suppression Kit Installed**

45. The objective of this evaluation was to determine if there was any appreciable degradation of UH-1H handling qualities with the FSK installed. Both static and dynamic tests were conducted with the FSK installed. The FSK was filled with water and was attached to the aircraft by the cargo hook for all tests, unless otherwise noted. Quantitative test results are presented in figures 77 through 80, appendix F. Time history data were not recorded since a continuous recording data system was not installed in the aircraft. All tests conducted in level flight at altitude were at an approximate $C_T$ of 0.0036. Autorotational tests were not conducted.

46. Increasing forward longitudinal cyclic was required to stabilize the aircraft as airspeed was increased from 41 to 96 knots calibrated airspeed (KCAS). Variations in lateral and directional controls, in this airspeed range, were not apparent to the pilot. All control margins were more than 10 percent for the airspeed range and conditions tested. The static trim characteristics are acceptable for missions requiring use of the FSK (HQRS 2).

47. The static directional stability and dihedral effect of the UH-1H helicopter with the FSK installed were generally positive (increasing right directional control, and left lateral control with increasing left sideslip) for the two trim airspeeds investigated. The variation in directional control requirements was essentially linear as sideslip was varied about trim. The lateral control displacement as a function of sideslip was nonlinear and became less positive as angle of sideslip was increased. This decrease in lateral cyclic control gradient indicated a decrease in dihedral effect. The static lateral-directional characteristics with FSK installed are satisfactory.

48. Side-force characteristics, as indicated by bank angle during steady sideslips, were qualitatively determined to be positive.

49. The handling qualities of the UH-1H in rearward flight with the FSK installed were similar to those discussed in paragraphs 39, 41, and 48.

50. The aircraft motions following a longitudinal control pulse input in hover, partial power descent, and level flight were well damped, and no undesirable cross-coupling motions were present. The aircraft reaction to longitudinal inputs was immediate and in the direction commanded. The motions of the FSK following a longitudinal pulse input were well damped, and these motions were not evident to the flight crew.

51. Initial aircraft response following a lateral control pulse input in hover, partial power descent, and forward flight were damped. However, approximately 2 to 3 seconds following a lateral pulse, the resulting oscillating motion of the FSK induced an aircraft rolling motion. The magnitude of the aircraft rolling motion increased as the size of the lateral pulse was increased. The most critical test condition was during partial power descent at an airspeed of approximately 55 KCAS. For this test condition, approximately ten lateral oscillations of the FSK were observed following a 1-inch left lateral pulse input. Aircraft rolling motions associated with FSK oscillations following a 1-inch left lateral control pulse input were not quite as noticeable in a hover as in descending flight. The aircraft rolling motions resulting from FSK oscillations following a left lateral pulse input required considerable pilot compensation to achieve adequate performance during partial power descent and hover (HQRS 5). Increasing airspeed in level flight caused the resulting oscillations of the FSK to decrease in number following a left lateral pulse input. The pilot effort required to provide adequate performance in level flight was minimal at airspeeds in excess of 40 KCAS for left lateral 1-inch inputs (HQRS 3). A 1-inch right lateral pulse input resulted in only three oscillations of the FSK during partial power descent and hovering flight. The aircraft response and resulting FSK oscillations to right lateral pulse inputs required minimal pilot effort to realize desired performance for all flight conditions tested (HQRS 3). No objectionable cross-axis coupling was encountered following lateral pulse inputs. Elimination of the aircraft rolling motions associated with FSK oscillations are desirable for improved operation and mission capabilities.

52. The UH-1H demonstrated heavy damping following a directional control pulse input. Positive dihedral effect (roll opposite direction of sideslip) was evident following the initial portion of the directional control input. This rolling motion excited the lateral oscillation of the FSK, however, minimal pilot compensation was required to achieve desired performance (HQRS 3).

53. Slow, coordinated turns with roll rates of 2 degrees per second (deg/sec) during climbs, descents, and level flight were evaluated at an airspeed of approximately 55 KCAS. The maximum bank angle investigated was 30 degrees, right and left. Pilot effort to control the helicopter in a roll was minimal (HQRS 3).

54. Abrupt, coordinated turns (10 deg/sec roll rate) were evaluated during climbs and descents at a speed of approximately 55 KCAS. The oscillating motions of the FSK following the initial lateral control input increased the pilot effort required to maintain the desired roll attitude during the turn when rolling from right to left with bank angle changes greater than 20 degrees (10 degrees right to 10 degrees left). The pilot compensation required for a 20-degree change in roll attitude and resulting turn was minimal. Larger bank angle changes required increased pilot effort.

55. Vibration characteristics were qualitatively evaluated throughout the flight envelope. The vibration characteristics for the conditions tested were acceptable, and no apparent change in aircraft vibration was noted with FSK installed when compared to the standard aircraft.

# CONCLUSIONS

## GENERAL

56. The following general conclusions were reached upon completion of the tail rotor performance test of the UH-1H helicopter:

a. The percentage of total engine power absorbed by the tail rotor varied nonlinearly when the left directional control margin was less than 10 percent (para 17).

b. The hovering performance is adversely affected by the apparent tail rotor stall when the left directional control margin is less than 10 percent (para 20).

c. The handling qualities are unacceptable during translational flight (paras 27 through 31).

d. An undamped lateral-directional oscillation was encountered at speeds greater than 25 knots during IGE right translational flight (para 33).

e. The maximum tail rotor power encountered during translational flight was 140 horsepower (para 35).

f. The maximum right lateral moment consistent with 10-percent remaining left lateral control in left sideward flight was 35,000 in.-lb (para 38).

g. Additional testing is required to determine the maximum acceptable right lateral moment with the rigging of the main rotor swashplate other than 2.0 degrees down, left (para 38).

h. The handling qualities were not significantly affected by FSK installation with the exception of aircraft reaction in roll to lateral oscillations of FSK (para 51).

i. Three deficiencies and one shortcoming were encountered during this program (paras 57 and 58).

## DEFICIENCIES AND SHORTCOMINGS AFFECTING MISSION ACCOMPLISHMENT

57. Correction of the following deficiencies appears essential for adequate mission accomplishment:

a. Insufficient longitudinal control within the approved gross-weight/cg envelope (para 28).

b. Insufficient directional control (para 32).

c. Directional instability between 10 and 18 knots at relative azimuths between 210 and 320 degrees is a safety-of-flight hazard (para 30).

58. Correction of the following shortcoming is desirable for improved operation and mission capabilities: aircraft rolling motions associated with FSK oscillations (para 53).

## MILITARY SPECIFICATION COMPLIANCE

59. All translational flight handling qualities requirements contained in MIL-H-8501A were complied with, except for the intent of the following paragraphs:

| Paragraph | Item |
|---|---|
| 3.2.1 | Insufficient longitudinal control (see para 28) |
| 3.3.2 and 3.3.6 | Insufficient directional control (see para 29) |

## RECOMMENDATIONS

60. Correct deficiencies prior to further procurement.

61. Correct shortcoming at earliest convenience.

62. Restrict the operational flight envelope to conditions which provide 10-percent longitudinal and directional control margins (para 32).

63. Limit the maximum right lateral moment to 35,000 in.-lb (para 38).

# APPENDIX A. REFERENCES


1. Final Report, USAASTA, Project No. 66-04, *Engineering Test, YUH-1H Helicopter, Phase D (Limited)*, November 1970.

2. Final Report, Air Force Flight Test Center, AFFTC-TR-64-27, *Category II Performance Tests of the YUH-1D with a 48-Foot Rotor*, November 1964.

3. Letter, ASD, ASD/SDQH 4-30, 12 April 1971, subject: HH-1H Tail Rotor Hover Performance.

4. Letter, ASD, ASD/SDQH 6-128, 22 June 1971, subject: UH-1H Tail Rotor Hover Performance.

5. Letter, AVSCOM, AMSAV-R-F, 11 June 1971, subject: Test Directive, UH-1H Tail Rotor Performance.

6. Letter, AVSCOM, AMSAV-R-F, 30 June 1971, subject: Test Directive, UH-1H Tail Rotor Performance.

7. Military Interdepartmental Purchase Request, USAFASC/PPPC, MIPR No. FX 2826-71-05336, 24 June 1971, subject: UH-1H Tail Rotor Hover Performance.

8. Military Interdepartmental Purchase Request, USAFASD/PPPC, MIPR No. FX 2826-71-05336, 8 July 1971, subject: UH-1H Tail Rotor Hover Performance, Amendment 1.

9. Technical Manual, TM 55-1520-210-10, *Operator's Manual, Army Model UH-1D/H Helicopter*, 7 May 1969, with Changes 1 and 2, 29 April 1970.

10. Technical Manual, TM 55-1520-210-20, *Organizational Maintenance Manual, Army UH-1D/H Helicopter*, 7 May 1969 with Changes 1 through 15, 22 June 1971.

11. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities, General Requirements For*, 7 September 1961, with Amendment 1, 3 April 1962.

12. Army Regulation, AR 310-25, *Dictionary of United States Army Terms*, 1 March 1969.

13. Report, Bell Helicopter Company, 805-099-400, *Basic Structural Design Criteria for UH-1D Utility Helicopter*, Revision F, 27 March 1969.

14. Final Report, USAASTA, Project No. 66-06, *Engineering Flight Test, AH-1G Helicopter, HueyCobra, Phase D, Part 1, Handling Qualities*, December 1970.

## APPENDIX B. BASIC AIRCRAFT INFORMATION AND OPERATING LIMITS

### AIRFRAME

#### Main Rotor System

1. The main rotor assembly is of the two-bladed, semirigid teetering type employing preconing and underslinging. The main rotor blades are all-metal bonded, and each blade is connected to a common yoke by means of a grip and suitable pitch change bearings with tension straps to carry centrifugal forces. The main rotor head (consisting of the yoke, blade grips, and bearings) is mounted to the mast by means of a trunnion through the teetering bearing. The trunnion permits rotor flapping while the blade grip to yoke extension bearings permit cyclic and collective pitch action. The main rotor control system consists of a swashplate assembly which transfers cyclic control motions from the fuselage-based system to the rotating controls, a scissors assembly which transfers motion from the rotating swashplate to the stabilizer bar mixing levers, a stabilizer bar which aids in the stability of the aircraft, a hydraulic damper assembly to control the "following time" of the stabilizer bar, and static stops to limit teetering motion of the hub.

#### Tail Rotor System

2. The tail rotor is a two-bladed, rigid, Delta-three hinged type employing preconing and underslinging. Each blade is connected to a common yoke by means of a grip and suitable pitch change bearings. The blade and yoke assembly is mounted on the tail rotor shaft by means of a Delta-three hinged trunnion to minimize rotor flapping. A pitch-change mechanism actuated by the tail rotor control pedals is provided to increase or decrease the pitch of the blades. The tail rotor system principal subassemblies are the rotor blades, each blade constructed of aluminum alloy; the tail rotor head consisting of blade grips, the yoke which forms the hub of the tail rotor, and the flapping axis trunnion which attaches the yoke to the shaft through the flapping axis bearing; and the tail rotor head blade pitch control mechanism, which consists of a push/pull tube that actuates a crosshead which is connected to the blade grips by means o control links to produce the desired blade pitch angles.

#### Empennage

3. The empennage consists of a vertical fin and synchronized elevator. The synchronized elevator, which has an inverted airfoil section, is located near the aft end of the tail boom and is connected by control tubes and mechanical linkage to the fore and aft cyclic control system. Fore and aft movements of the cyclic control stick produce a change in the synchronized elevator attitude. The swept-back vertical fin extends up from the aft end of the tail boom and houses a portion of the tail rotor drive shaft. The vertical tail has no control surfaces.

friction can be induced into the control lever by hand tightening the friction adjuster. A rotating grip-type throttle and a switch-box assembly are located on the upper end of the pilot collective control pitch lever. The copilot collective pitch control lever contains only the rotating grip-type throttle, starter switch, and governer rpm increase/decrease switch.

### Tail Rotor Pitch Control Pedals

9. Tail rotor pitch control pedals alter the pitch of the tail rotor blades, and thereby provide the means for directional control. The force trim system is connected to the directional controls and is operated by the force trim switch on the cyclic control grip.

## ENGINE

### Engine Description

10. The T53-L-13 engine, rated at 1,400 shp, is a free-turbine-type power plant. The main subassemblies of the engine are an inlet section, compressor section, diffuser section, combustor section, and exhaust section. The engine is derated to 1,100 shp because of airframe drive train torque limits. All sections are designed to include an annular flow path for the air or hot gases, are structurally interdependent, support all internal rotating systems, and provide attaching capabilities for engine-required external components and limited airframe accessories.

### Engine Power Control System

11. The T53-L-13 engine has a hydromechanical fuel control which consists of the following main units:

a. Dual-element fuel pump.

b. Gas producer speed governor.

c. Power turbine speed topping governor.

d. Acceleration and deceleration control.

e. Fuel shut-off valve.

f. Transient air bleed control.

12. An air bleed control is incorporated within the fuel control to provide for opening and closing the compressor interstage air bleed in response to the following signals present in the fuel control:

## Transmission System

4. The transmission is mounted forward of the engine and is coupled to the engine by a short drive shaft. The transmission is basically a reduction gearbox which transmits engine power at reduced rpm to the main and tail rotors by means of a two-stage planetary geartrain. The transmission incorporates a free-wheeling unit at the input drive which provides a disconnect from the engine in case of a power failure and allows the aircraft to autorotate. The tail rotor is powered by a takeoff on the lower aft section of the transmission.

## Control Systems

5. The flight control system is a positive mechanical type actuated by conventional helicopter controls. The system includes a cyclic control stick, the collective pitch (main rotor) control lever, tail rotor (directional) control pedals, and synchronized elevator connected mechanically to the fore and aft cyclic control system.

## Force Trim

6. Force centering devices are incorporated in the cyclic controls and directional pedal controls. These devices are installed between the cyclic stick and the hydraulic servo cylinders, and between the directional control pedals and the hydraulic servo cylinder. These devices furnish a force gradient to the cyclic stick and directional control pedals. The force trim can be deactuated by keying the left button on the top of the cyclic stick or by cycling the force trim ON/OFF switch installed on the hydraulic control panel to the OFF position. The gradient is accomplished by springs and magnetic brake release assemblies which enable the pilot to trim the controls as desired.

## Cyclic Pitch Control Stick

7. The cyclic pitch control stick operates the longitudinal and lateral control systems of the aircraft. The synchronized elevator is linked to the fore and aft cyclic stick movements by means of mechanical linkage and connecting control tubes. The pilot cyclic stick grip contains the cargo release switch; a trigger-type, three-position radio transmitter switch; armament fire control switch; hoist switch; and the force trim release switch. Desired pilot cyclic control operating friction can be induced by hand tightening a friction adjuster. The copilot cyclic control stick is the same as the pilot cyclic control stick, with the exception that the copilot stick does not have a friction adjuster.

## Collective Pitch Control

8. The collective pitch control levers are located to the left of the pilot and copilot, respectively. Main rotor blade pitch is controlled by this lever. When the lever is in the full-down position, the main rotor is at minimum pitch. When the lever is in the full-up position, the main rotor is at maximum pitch. Operating

a. Gas producer speed.

b. Compressor inlet air temperature.

c. Fuel flow.

13. The fuel control is designed to be operated either automatically or in an emergency mode. In the emergency position, fuel flow is routed around the main metering valve to the manual (emergency) metering and dump valve assembly. While in the emergency mode, fuel flow to the engine is controlled by the position of the manual metering valve which is directly connected to the power control (twist grip). During the emergency operation, there is no automatic control of fuel flow during acceleration and deceleration, thus engine exhaust gas temperature (EGT) and engine acceleration must be pilot monitored.

## BASIC AIRCRAFT INFORMATION

### Airframe Data

| | |
|---|---|
| Overall length (main rotor fore and aft and tail rotor horizontal) | 684.67 in. |
| Overall width (rotor trailing) | 114.6 in. |
| Center line of main rotor to center line of tail rotor | 345.9 in. |
| Center line of main rotor to elevator hinge line | 246.5 in. |
| Elevator area (including protected area of tail boom | 23.7 $ft^2$ |
| Elevator area (both panels) | 19.8 $ft^2$ |
| Elevator airfoil section | Clark Y (inverted) |
| Vertical stabilizer area | 11.3 $ft^2$ |
| Vertical stabilizer airfoil section | NACA 0015 |
| Vertical stabilizer aerodynamic center | FS 443.9, WL 112.1 |

### Main Rotor Data

| | |
|---|---|
| Number of blades | 2 |
| Diameter | 48 ft |

| | |
|---|---|
| Disc area | 1,809 $ft^2$ |
| Blade chord | 21 in. |
| Rotor solidity | 0.0464 |
| Blade area (both blades) | 84 $ft^2$ |
| Blade airfoil | NACA 0012 |
| Linear blade twist (root to tip) | -10 deg |
| Hub precone angle | 2.75 deg |
| Mast angle (relative to horizontal reference) | 5 deg forward tilt |
| Test aircraft control travel: | |
| Collective (measured at center of grip) | 11.0 in. |
| Longitudinal cyclic (measured at center of grip) | 12.9 in. |
| Lateral cyclic (measured at center of grip) | 12.6 in. |
| Directional (measured at center of pedal) | 6.9 in. |
| Blade travel: | |
| Flapping (any direction) | ±11 deg |
| Longitudinal cyclic | +12 to -12 deg |
| Lateral cyclic (rigged 2 deg down, left) | +9 to -11 deg |

<u>Antitorque Rotor Data</u>

| | |
|---|---|
| Number of blades | 2 |
| Diameter | 8.5 ft |
| Disc area | 56.7 $ft^2$ |
| Blade chord | 8.41 in. |
| Rotor solidity | 0.105 |
| Blade airfoil | NACA 0015 |

| | |
|---|---|
| Blade twist | Zero deg |
| Blade travel (average): | |
| Full left pedal | 18 deg |
| Full right pedal | -10 deg |

<u>Gross-Weight/Center-of-Gravity Envelope</u>

Forward cg limit:

Below 8,600 pounds, FS 130.0; linear increase from 8,600 pounds, FS 130.0, to FS 134.0 at 9,500 pounds.

Aft cg limit:

Below 8,600 pounds, FS 144.0; linear decrease from 8,600 pounds, FS 144.0, to FS 143.0 at 9,500 pounds.

<u>Rotor and Engine Speed Limits</u> (Steady State)

| | |
|---|---|
| Power on: | |
| Engine rpm | 6,400 and 6,600 |
| Rotor rpm | 314 and 324 |
| Transient rpm | 331 |
| Power off: | |
| Rotor rpm | 294 and 339 |

<u>Temperature and Pressure Limits</u>

| | |
|---|---|
| Engine oil temperature | 93°C |
| Transmission oil temperature | 110°C |
| Engine oil pressure | 25 to 100 psi |
| Transmission oil pressure | 30 to 70 psi |
| Fuel pressure | 5 to 20 psi |

## Gear Ratios

| | |
|---|---|
| Power turbine to engine output shaft | 3.2105:1 |
| Engine output shaft to main rotor | 20.370:1 |
| Engine output shaft to antitorque rotor | 3.990:1 |
| Engine output shaft to antitorque drive system | 1.535:1 |
| Gas producer turbine to tachometer pad (100% = 25, 150 rpm) | 5.988:1 |

## Engine and Drive Train Limits

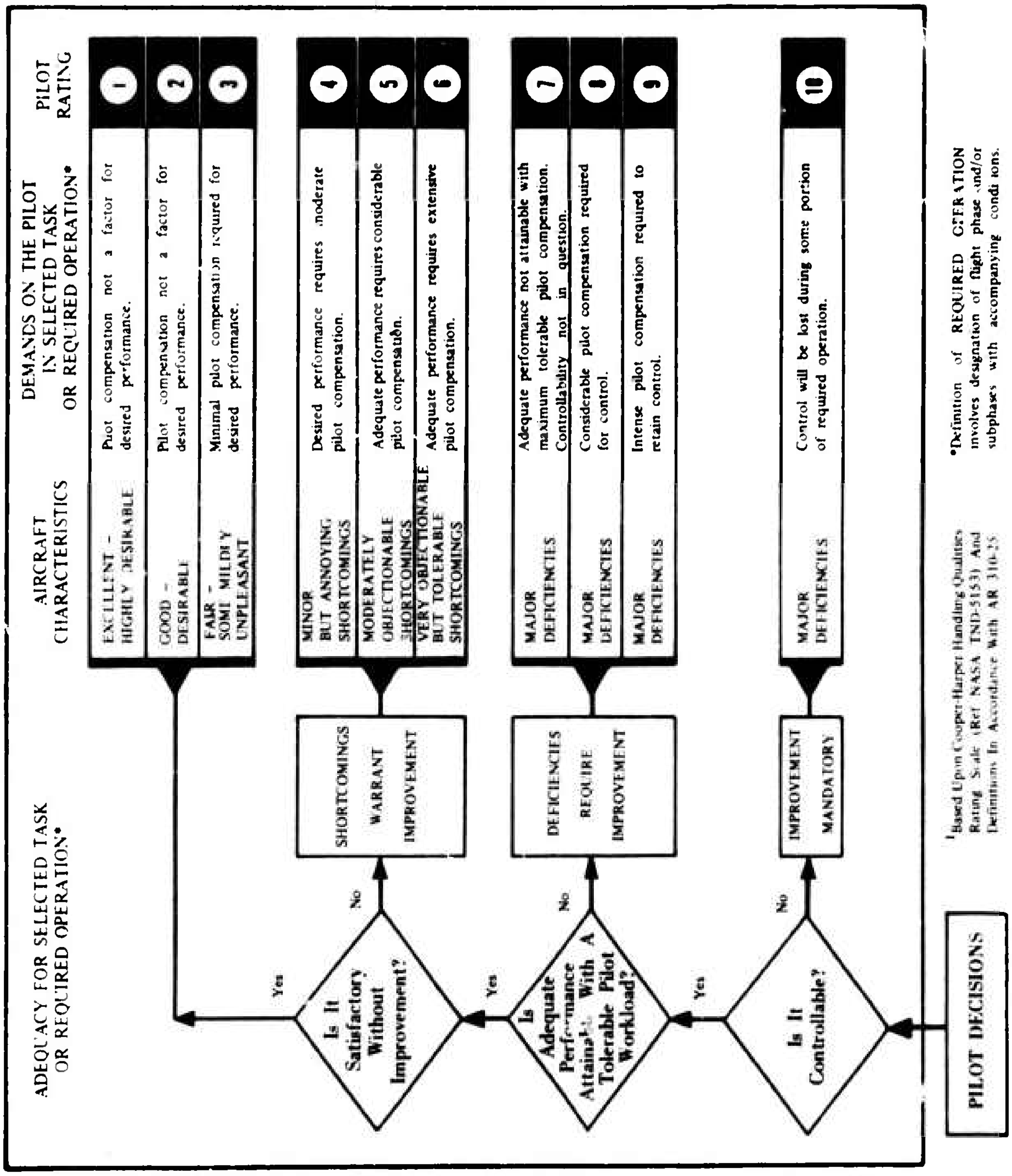
| | |
|---|---|
| Power ratings: | |
| Military power (30-minute limit) | 1,400 shp derated to 1,100 shp |
| Maximum continuous power | 1,250 shp derated to 1,100 shp |
| Torque limits: | |
| Maximum continuous | 50 psi |
| Transient overtorque (not to be used intentionally) (no maintenance required) | 50 to 54 psi |
| Transient overtorque (inspect drive train) | 54 to 61 psi |
| Transient overtorque (replace all drive train and rotor components) | Over 61 psi |
| Output shaft speed: | |
| Maximum steady state | 6,600 rpm |
| Minimum steady state | 6,400 rpm |
| Minimum steady state below 7,500 pounds | 6,000 rpm |
| Maximum transient (not to be used intentionally) | 6,750 rpm |

| | |
|---|---|
| **Exhaust Gas Temperature** | |
| Maximum continuous | 390°C to 625°C |
| 30-minute limit | 625°C to 645°C |
| 5-second limit for starting and acceleration | 675°C |
| Maximum for starting and acceleration | 760°C |
| **Gas Producer** | |
| Maximum speed | 25,600 rpm (101.8 percent) |
| Flight idle speed | 15,900 to 17,000 rpm (63 to 68 percent) |
| Ground idle/start speed | 12,100 to 13,100 rpm (48 to 52 percent) |
| **Airframe** | |
| Loading: | |
| Design weight | 6,600 lb |
| Maximum overload weight | 9,500 lb |
| Maximum floor loading | 300 lb/ft$^2$ |
| Maximum cargo hook capacity | 4,000 lb |
| Maximum lateral cg | Consult report for recommendations |
| Limit load factors: | |
| Positive 6,600 lb | +3.0g's |
| Positive 9,500 lb | +2.1g's |
| Negative 6,600 lb | -0.5g |
| Negative 9,500 lb | -0.35g |

Maximum airspeed:

| | |
|---|---|
| Forward flight | 124.0 KTAS at 2,000 ft |
| Sideward and rearward flight | Consult report for recommendations |

# APPENDIX C. HANDLING QUALITIES RATING SCALE



## APPENDIX D. TEST TECHNIQUES AND DATA REDUCTION PROCEDURES

### INTRODUCTION

#### Nondimensional Method

1. The helicopter performance results may be generalized through use of nondimensional coefficients. The test results obtained at specific test conditions may be used to accurately define performance at conditions not specifically tested. The following nondimensional coefficients were used to generalize test resu'ts obtained during this test program:

$$\text{Power Cofficient} = C_P = \frac{550\ \text{SHP}}{\rho A\ (\Omega R)^3} \tag{1}$$

$$\text{Thrust Coefficient} = C_T = \frac{\text{GRWT}}{\rho A\ (\Omega R)^2} \tag{2}$$

$$\text{Tip-Speed Ratio} = \mu = \frac{1.689\ V_T}{\Omega R} \tag{3}$$

$$\text{Main Rotor Advancing Tip Mach Number} = M_{tip}$$

$$= \frac{1.689\ V_T + \Omega R}{a} \tag{4}$$

2. Correlation of handling qualities was accomplished by summarizing the quantitative data as a function of main rotor thrust coefficient ($C_T$). Each individual handling qualities test flight was flown at a constant $C_T$. A constant $C_T$ was maintained by either increasing altitude as fuel was consumed (for flights conducted at altitude) or adding ballast to the aircraft as fuel was consumed (for flights conducted IGE).

#### Instrumentation

3. All instrumentation was calibrated prior to commencing the test program. All quantitative data obtained during this flight test program were derived from special sensitive instrumentation. A list of the instrumentation is given in appendix E. Data were obtained from three aircraft sources and three ground sources. The

aircraft sources were the engineer test panel, the copilot panel, and the pilot panel. All data from the aircraft were transmitted by radio and were hand recorded on the ground. A hand-held tape recorder was carried in the aircraft as a backup system in the event of radio malfunction. The ground support sources were a load cell (used for hover tests), a ground weather station (used for hover and all IGE handling qualities tests), and a calibrated pace vehicle (used for all IGE handling qualities tests).

### Weight and Balance

4. The test aircraft was weighed prior to the installation of test instrumentation and was reweighed twice after test instrumentation was installed with the aircraft battery located in two positions: FS 5.0 (forward) and FS 233.0 (aft). The fuel load for each test flight was determined prior to engine start and after engine shutdown by measuring the fuel specific gravity and temperature, and by using an external calibrated sight gage connected to the fuel cells to determine total fuel volume. Fuel used in flight was recorded by a calibrated fuel-used system, and the final fuel-used reading following engine shutdown was cross-checked with the sight gage readings following each flight. Helicopter loading and cg (both lateral and longitudinal) were controlled by ballast installed at various locations in the aircraft.

## PERFORMANCE

### Antitorque System Performance

5. The performance of the antitorque rotor system in hover and translational flight was defined by measuring the parameters necessary to define tail rotor horsepower, tail rotor thrust, and directional control (pedal) position were measured. Tail rotor thrust was not determined for translational flight conditions.

6. Antitorque system output torque was measured at the output shaft of the 90-degree tail rotor gearbox. This torque was used to determine tail rotor horsepower by the following equation:

$$SHP_{TR} = TRQ_{TR} \times N_{TR} \times \frac{2\pi}{33,000} \quad (5)$$

7. The nondimensional tail rotor power coefficient was determined by the following equation:

$$C_{P_{TR}} = \frac{SHP_{TR} \times 550}{\rho A_{TR} \, (\Omega_{TR} \, R_{TR})^3} \quad (6)$$

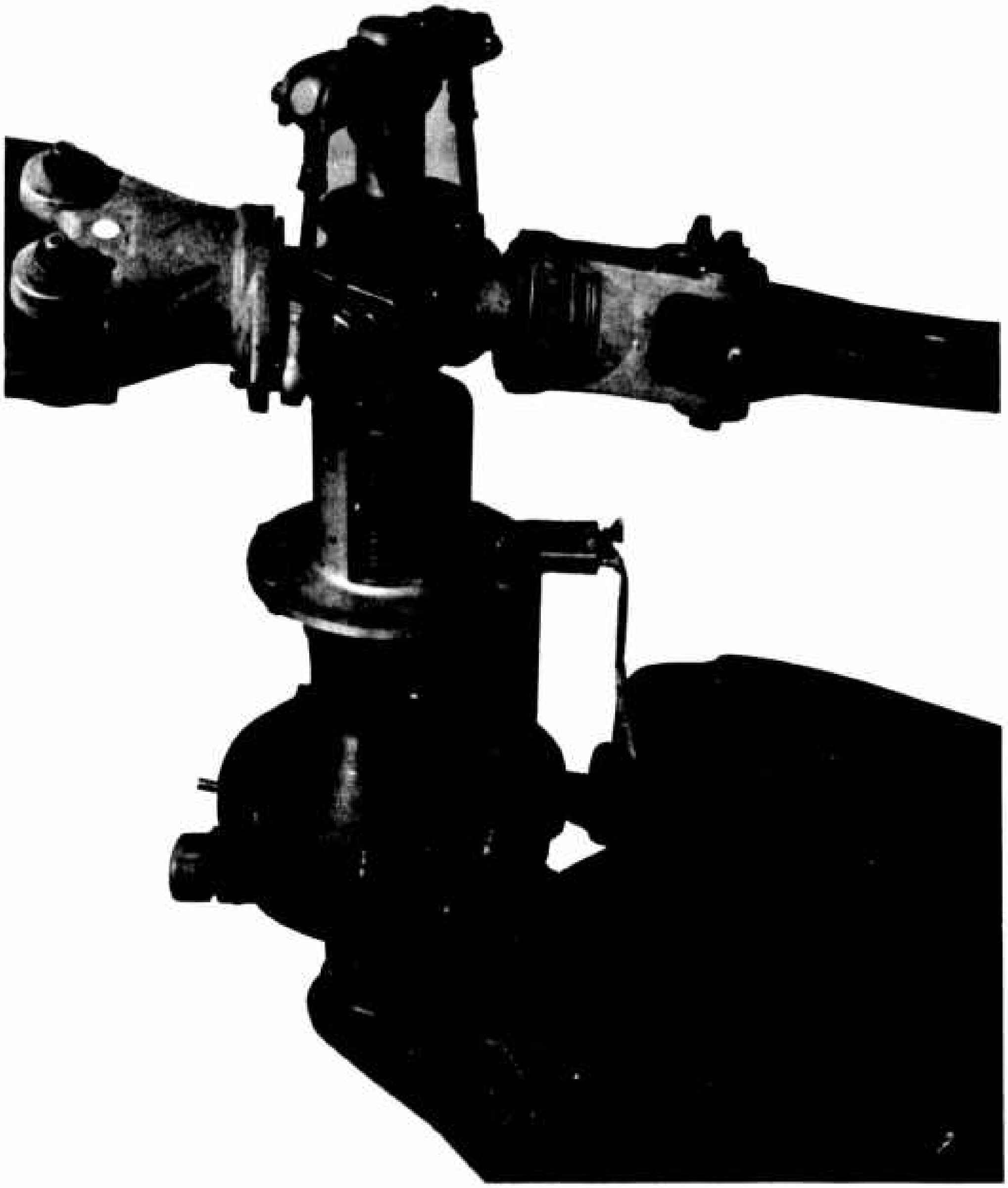


Photo A. Tail Rotor Slip-Ring Installation.

8. The tail rotor thrust for hover was determined by first making several assumptions: (1) All restoring directional moment to maintain stabilized hover was assumed to be generated by the antitorque system. This assumption neglected to consider any restoring directional moment which could be derived from rotor downwash and recirculating air flow over the fuselage, tail boom section, and/or vertical stabilizer; (2) Total power loss, attributed to frictional losses (gears, bearings, etc.) and power extracted from main transmission to drive accessories (hydraulic pumps), was assumed to be 5 percent of the engine output shp. This assumption was necessary to determine the horsepower delivered to the main rotor; and (3) This analysis further assumed that the free air temperature of the air mass flow passing through the tail rotor was not influenced by the hot gases being emitted from the engine.

9. The horsepower to the main rotor (MR) was determined by the following equation:

$$SHP_{MR} = SHP_{ENG} - SHP_{TR} - (0.05 \times SHP_{ENG}) \tag{7}$$

10. The nondimensional power coefficient of the main rotor was determined by the following equation:

$$C_{P_{MR}} = \frac{SHP_{MR} \times 550}{\rho A \; (\Omega R)^3} \tag{8}$$

11. The thrust from the tail rotor in a hover can be determined by the following equation:

$$THRUST_{TR} = \frac{TRQ_{MR}}{l_t} = \frac{550 \; SHP_{MR}}{\Omega_{MR} \; l_t} \tag{9}$$

12. Equation 9 was expanded to obtain the nondimensional thrust coefficient of the tail rotor:

$$C_{T_{TR}} = \frac{C_{P_{MR}} \; R \; A \; (\Omega R)^2}{l_t \; A_{TR} \; (\Omega_{TR} \; R_{TR})^2} \tag{10}$$

13. The position of the directional control was determined by measuring pedal position. Full left directional control application resulted in an average tail rotor

blade angle of 18 degrees for the test aircraft. The total directional control (pedal) displacement (full left to full right) resulted in a 28.6-degree change in tail rotor blade angle.

14. The nondimensional tail rotor performance and directional control position were used to determine tail rotor horsepower and directional control margins as a function of skid height. All antitorque data were obtained simultaneously with hover and translational flight tests.

## Hover

15. The tethered hovering technique was used to define hover performance. Various lengths of an intermediate cable, between the aircraft and load cell, were used to control the skid height of the helicopter. One end of the intermediate cable was attached to the helicopter by the cargo hook at FS 138.0, and the other was attached to the load cell. The load cell, used to measure cable tension, was secured to a ground by using a tie-down. For each skid height, the engine power was varied incrementally from a power that yielded approximately 300 pounds of cable tension to maximum power available. Prior to recording the data, the aircraft was stabilized with respect to vertical alignment, power, and control positions. When the power and cable tension were stabilized, the parameters necessary to define gross weight, cable tension, engine shp, and ambient air conditions were recorded. The cable tension was continuously monitored during each data point to ensure that a reasonable static condition was present while all other parameters were recorded. All hovering performance tests were conducted in less than 2 knots of wind.

16. Hovering data collected in terms of gross weight, shp, and ambient air conditions were converted to define the relationship between $C_T$ and $C_P$. This relationship was unique for each skid height. Summary hovering performance was calculated from nondimensional hovering curves by dimensionalizing the curves at selected ambient conditions.

## Level Flight

17. Level flight performance with the FSK installed was defined by measuring the shp required to maintain level flight as speed was varied. An almost constant $C_T$ was maintained by increasing altitude as fuel was consumed. Only one level flight performance test was conducted to determine the approximate increase in equivalent flat plate area with the FSK installed. The results of the level flight performance test were converted to nondimensional form. Nondimensional level flight test results were then compared to the level flight performance data presented in reference 1, appendix A, to determine the increase in flat plate area. Increase in equivalent flat plate area was calculated by the following equation:

$$\Delta f = \frac{2\ \Delta C_P\ A\ (\Omega R)^3}{(V_T \times 1.689)^3} = \frac{2\ \Delta C_P\ A}{\mu^3} \qquad (11)$$

## Power Determination

18. Engine power output, in terms of torque, was determined by measuring the engine torque effort in the cockpit on gages. The torquemeter system is essentially a piston (restrained by oil) that senses a pressure which is proportional to the power output of the engine. The observed engine pressure is converted to torque (in.-lb) by use of the engine acceptance test data. The results of the acceptance tests for the engine used during this evaluation are presented in figure I. This plot was used to obtain engine output torque. The engine torque range during these tests was not sufficient to cover the entire operating torque range. Engine horsepower data obtained during this program correlated very well with previous test results which employed calibrated T53-L-13 engines.

19. Horsepower transmitted by a rotating shaft may be expressed in the following manner:

$$SHP = \frac{2\pi}{33{,}000} \times N_E \times TRQ \qquad (12)$$

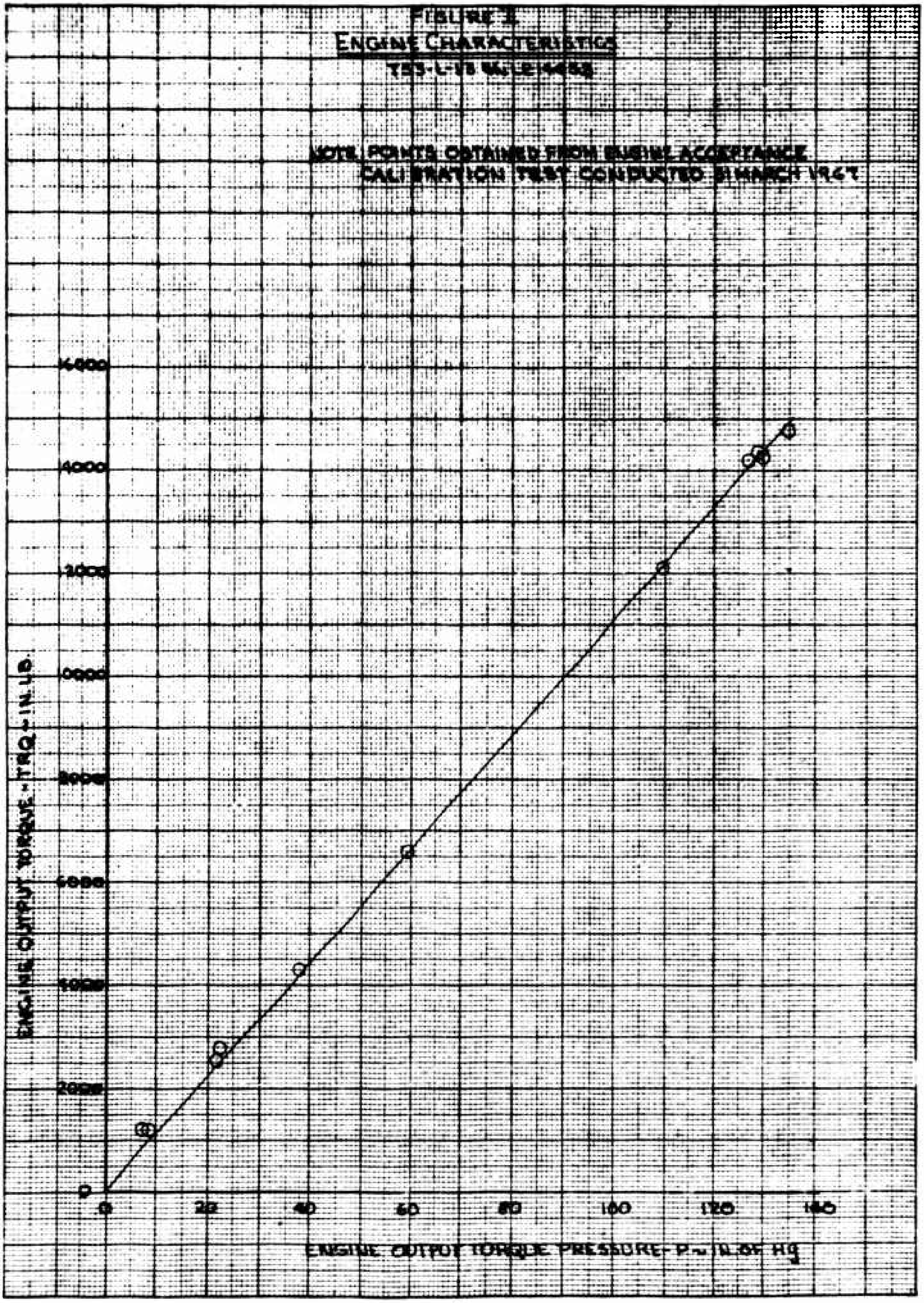
20. Engine output shaft speed was determined from rotor speed by using the following:

$$N_E = N_R \times 20.370 \qquad (13)$$

21. Substituting equation 13 into equation 12, a convenient equation for determining output shp can be developed:

$$SHP = \frac{2\pi \times 20.383 \times TRQ \times N_R}{33{,}000} = 3.861 \times 10^{-3} \times TRQ \times N_R \qquad (14)$$

22. This equation was used during the program to determine the shp for each test condition.


ENGINE CHARACTERISTICS
ENGINE OUTPUT TORQUE - TRQ - IN. LB.
ENGINE OUTPUT TORQUE PRESSURE
0
20
40
60
80
100
120
140
2000
4000
6000
8000
10000
12000
14000
16000

## HANDLING QUALITIES

### Flight Control Systems

23. The limits of the cyclic control pattern were measured on the ground with the rotor in a static position. Hydraulic pressure and electrical power were supplied by ground support equipment during this test. The cyclic control was rotated around the boundary every 5 percent of longitudinal or lateral control. The position of the longitudinal and lateral cyclic control were both read at each 5-percent value.

24. Tail rotor blade angle, as a function of directional control position, was measured on the ground with the tail rotor in a static position. Hydraulic pressure and electrical power were again supplied by ground support equipment. The tail rotor blade angle was determined by measuring the blade angle of each blade at every 5-percent increment of directional control displacement. The two measured blade angle values were then added algebraically and divided by two. The test was conducted both with the directional control being displaced left to right, and vice versa, to determine the amount of hysteresis in the control system.

### Static Trim Stability

25. The static trim stability was investigated by trimming the helicopter at various airspeeds over an airspeed range. While the aircraft was stabilized at each trim airspeed, all control positions were recorded. Altitude was varied during each test flight to maintain a constant thrust coefficient for each trim airspeed.

### Static Directional Stability and Effective Dihedral

26. The static directional stability and effective dihedral tests were conducted using the following technique. The helicopter was first stabilized at a trim airspeed with the ball centered on the turn and bank indicator. The yaw attitude was then varied and stabilized at different values, while the magnetic heading of the flight path was held constant. All control positions were recorded at each stabilized yaw attitude angle. Altitude was varied as fuel was consumed during each test to maintain a constant thrust coefficient.

### Translational Flight Evaluation

27. The translational handling qualities were investigated by conducting tests at various combinations of wind azimuth and airspeed. When the aircraft was stabilized in translational flight, parameters necessary to determine gross weight, ambient air conditions, azimuth, airspeed control positions, and tail rotor horsepower were recorded. A ground vehicle with a calibrated speedometer was used as a reference when attempting to stabilize the helicopter at the desired airspeed and azimuth. Ambient wind velocity and direction were incorporated into the analysis when determining the airspeed and wind azimuth. Tests were conducted with wind velocties less than 4 knots. A constant thrust coefficient was maintained for each test condition by adding ballast as fuel was consumed.

## Sideward and Rearward Flight

28. The test method and parameters recorded during sideward and rearward flight were the same as translational flight evaluation, with one exception. The azimuth headings investigated were limited to 90, 180, and 270 degrees.

## Dynamic Stability

29. Dynamic stability characteristics of the UH-1H were tested by using the following techniques. The aircraft was first trimmed at the desired flight condition and airspeed. Gust disturbances were then simulated by making pulse-type control inputs of 1 inch for 0.5 to 1.0 second. The control was then returned to trim at which time all controls were held fixed until the aircraft motions damped out or recovery action was required. Qualitative comments were made during and after each pulse-type control input.

## APPENDIX E. TEST INSTRUMENTATION

All instrumentation was calibrated and installed prior to commencing the test program. All quantitative data obtained during this flight test program were derived from special sensitive instrumentation and were hand recorded. Data were obtained from three aircraft sources and three ground support sources, two of which were used at all times, depending on the type of test involved. The aircraft sources were the pilot panel, the copilot panel, and the engineer panel. The ground support sources were a load cell (used for hover tests), a ground weather station (used for hover and translational flight), and a calibrated pace vehicle (used for translational flight evaluation). All data from the aircraft were transmitted by radio and were hand recorded on the ground. A hand-held tape recorder was carried in the aircraft as a backup system in the event of radio malfunction. A detailed tabulation of the instrumentation is given below:

**PILOT PANEL**

Rotor speed

**COPILOT PANEL**

High torque
Low torque
Altimeter

**ENGINEER TEST PANEL**

Tail rotor torque
Fuel counter
Longitudinal stick position
Lateral stick position
Directional control position
Collective control position

**LOAD CELL**

Cable tension

**GROUND WEATHER**

Free air temperature
Altimeter

**PACE VEHICLE**

Calibrated fifth wheel

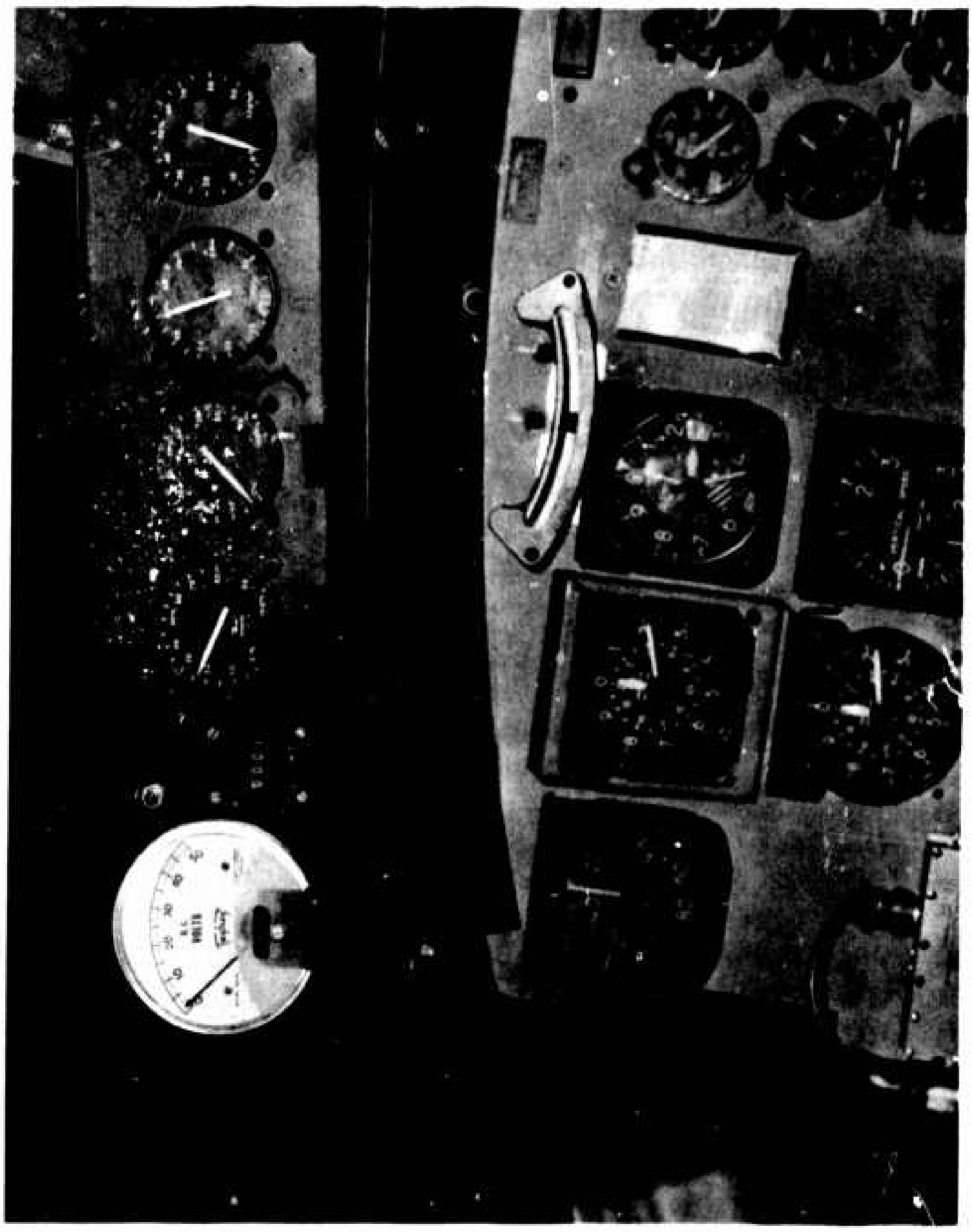


Photo I. Engineer Test Panel.

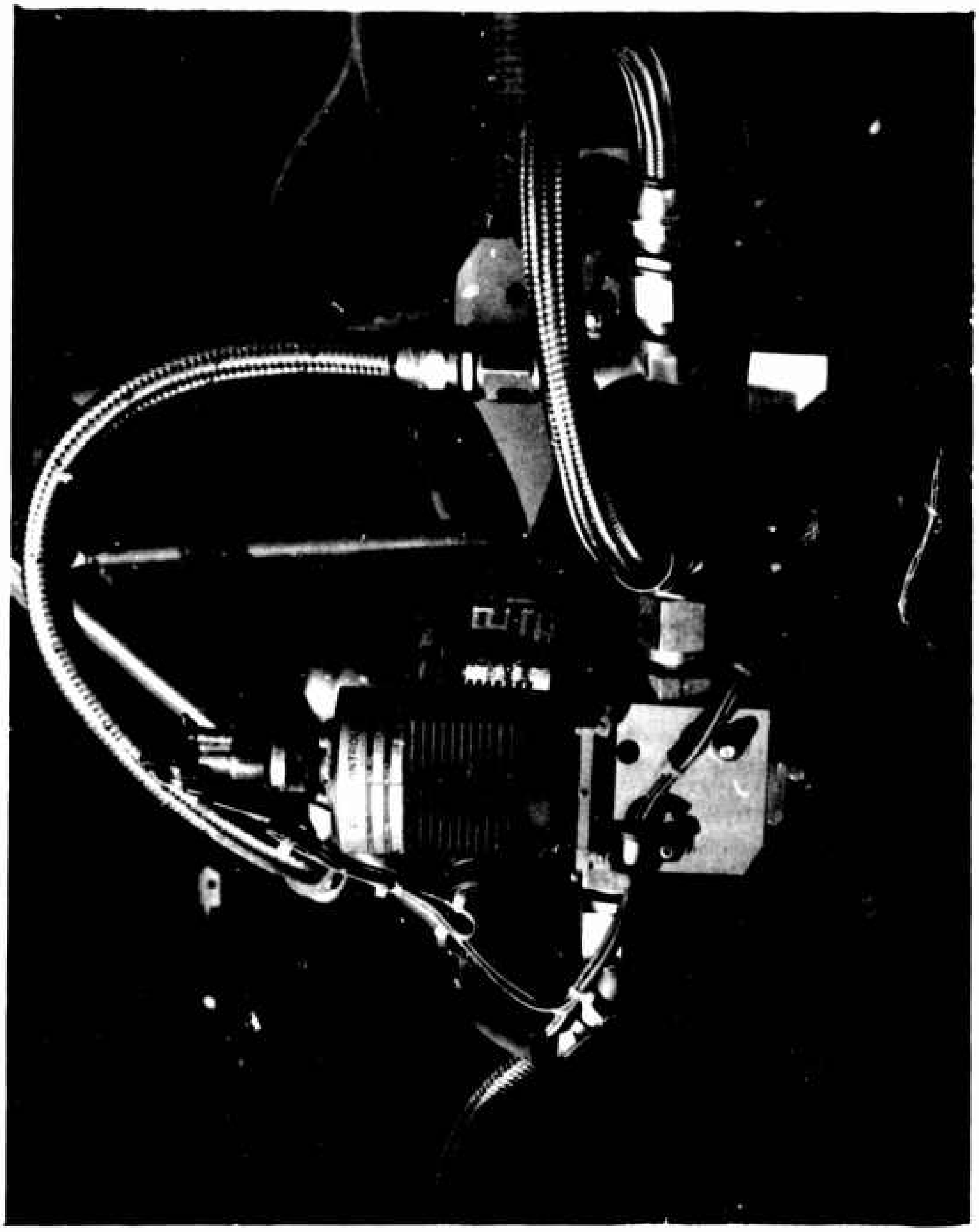


Photo II. Fuel Flow Meter with By-Pass Valve.

## APPENDIX F. TEST DATA

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| Average Tail Rotor Blade Angle versus Directional Control Position | 15 |
| IGE Hovering Performance | 16 |
| Nondimensional Hovering Performance | 17 through 23 |
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| Cyclic Pitch Control Pattern | 25 |
| IGE Hovering in Wind for a 10-percent Control Margin of all Controls | 26 |
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| Right Lateral CG Investigation in Rearward Flight | 74 through 76 |
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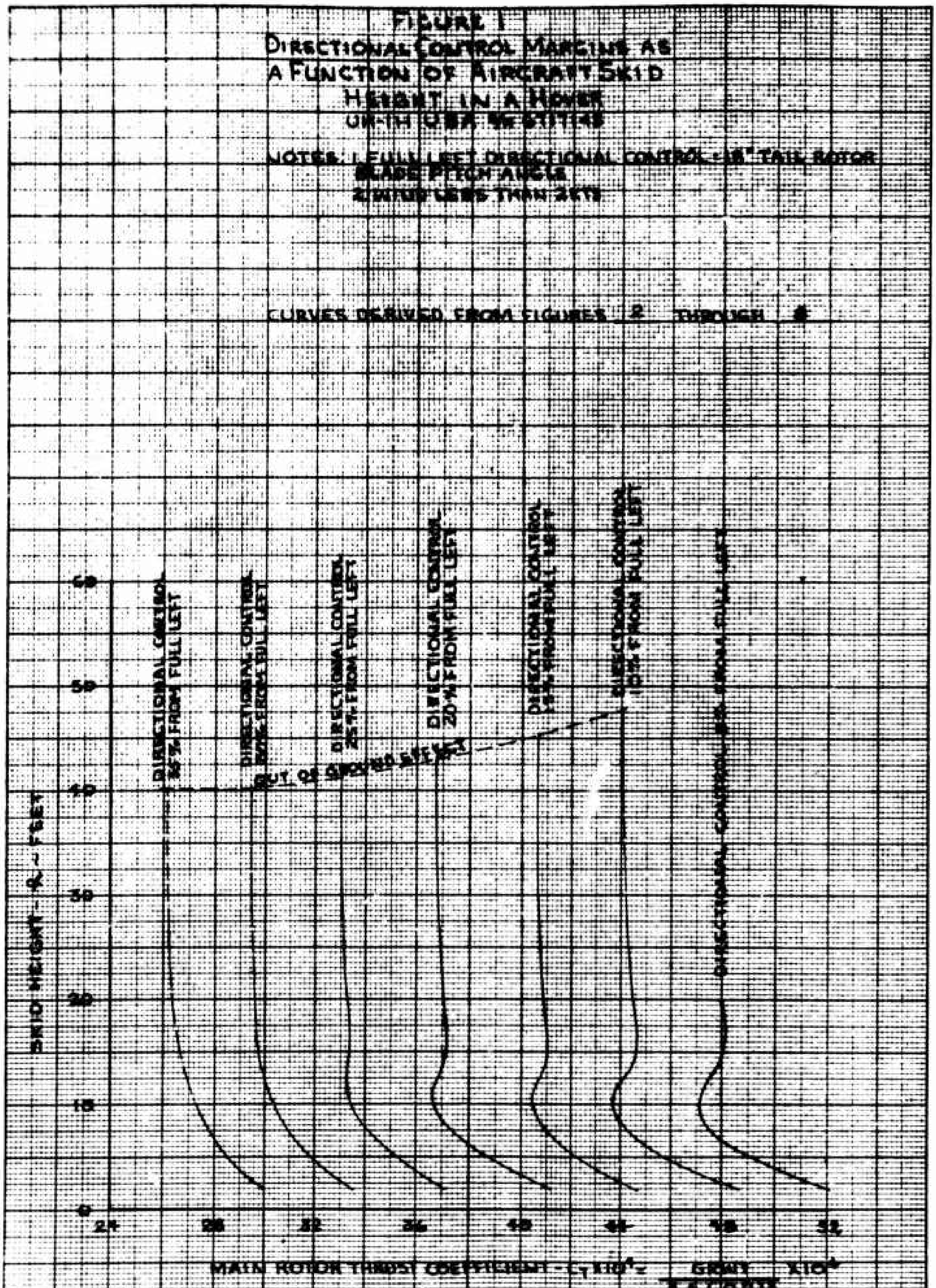


FIGURE 1
DIRECTIONAL CONTROL MARGINS AS
A FUNCTION OF AIRCRAFT SKID
HEIGHT IN A HOVER
NOTES: 1. FULL LEFT DIRECTIONAL CONTROL
BLADE PITCH ANGLE
CURVES DERIVED FROM FIGURES
THROUGH
DIRECTIONAL CONTROL
FROM FULL LEFT
25% FROM FULL LEFT
20% FROM FULL LEFT
10% FROM FULL LEFT
OUT OF GROUND EFFECT
SKID HEIGHT
FEET
0
10
20
30
40
50
60
24
28
32
36
40
44
48
52
MAIN ROTOR THRUST COEFFICIENT

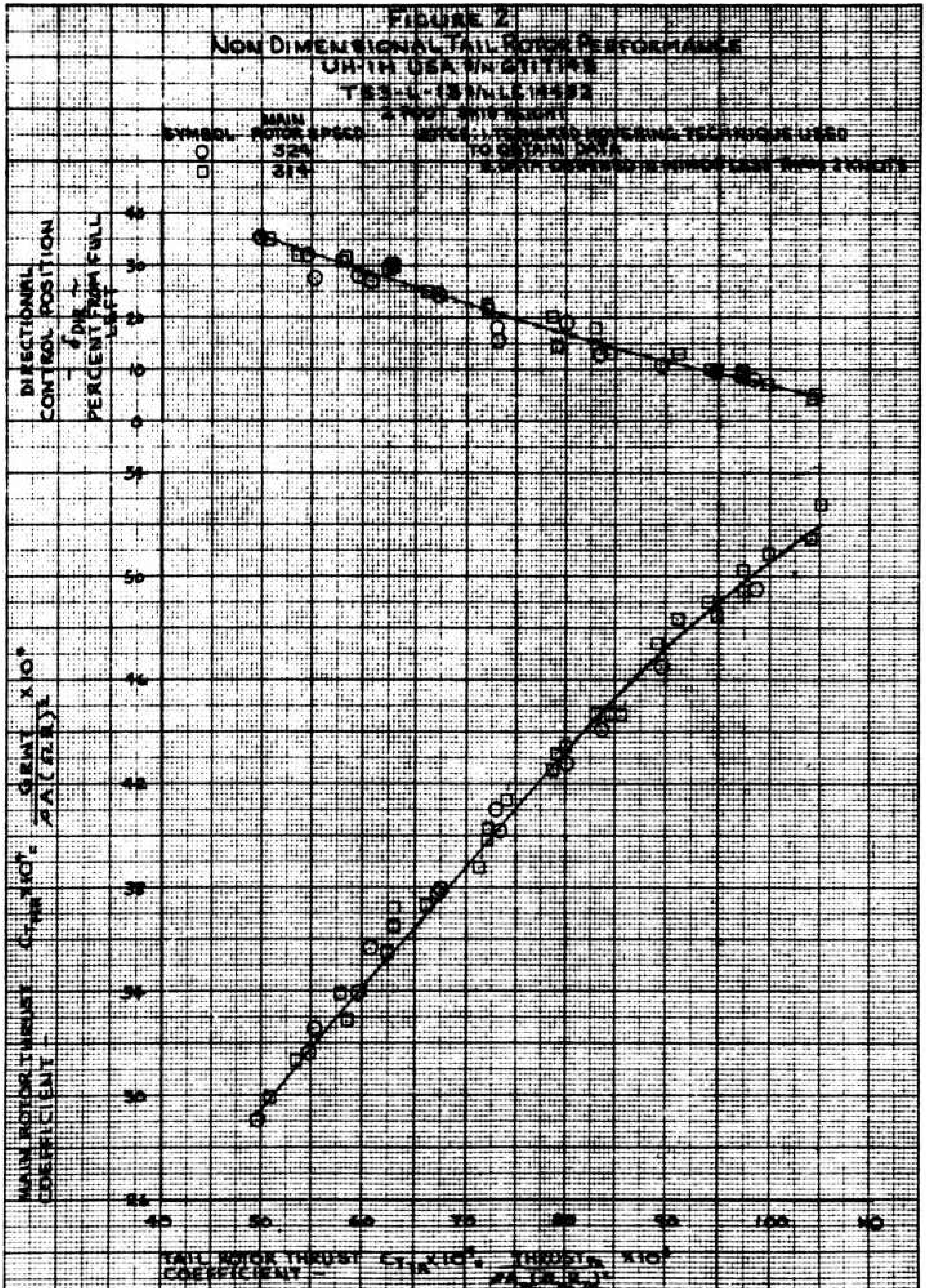
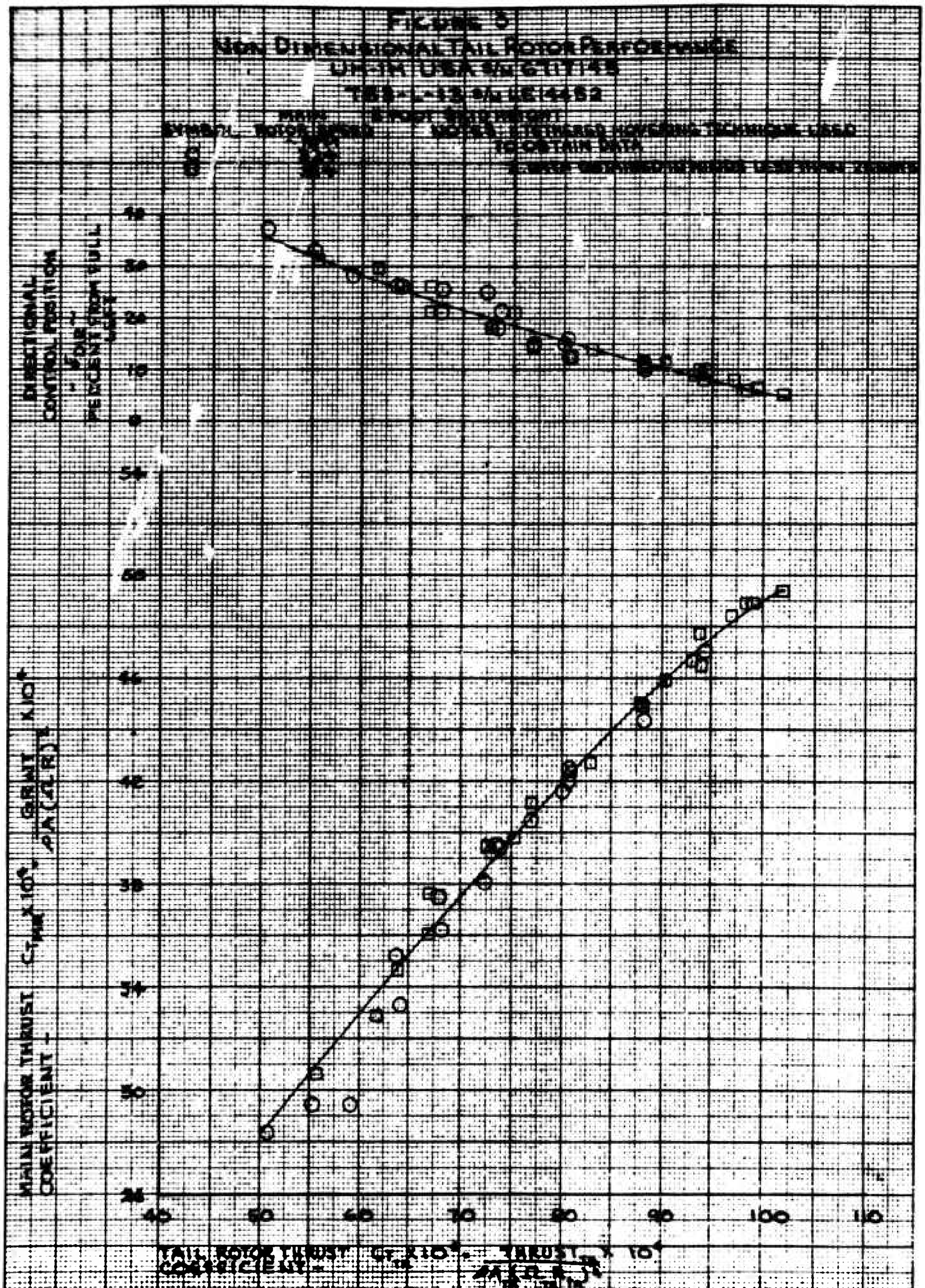


FIGURE 2
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
SYMBOL
MAIN ROTOR SPEED
324
314
DIRECTIONAL CONTROL POSITION
PERCENT FROM FULL LEFT
MAIN ROTOR THRUST COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
40
50
60
70
80
90
100
110



NON DIMENSIONAL TAIL ROTOR PERFORMANCE
TO OBTAIN DATA
DIRECTIONAL CONTROL POSITION
MAIN ROTOR THRUST COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
40
50
60
70
80
90
100
110

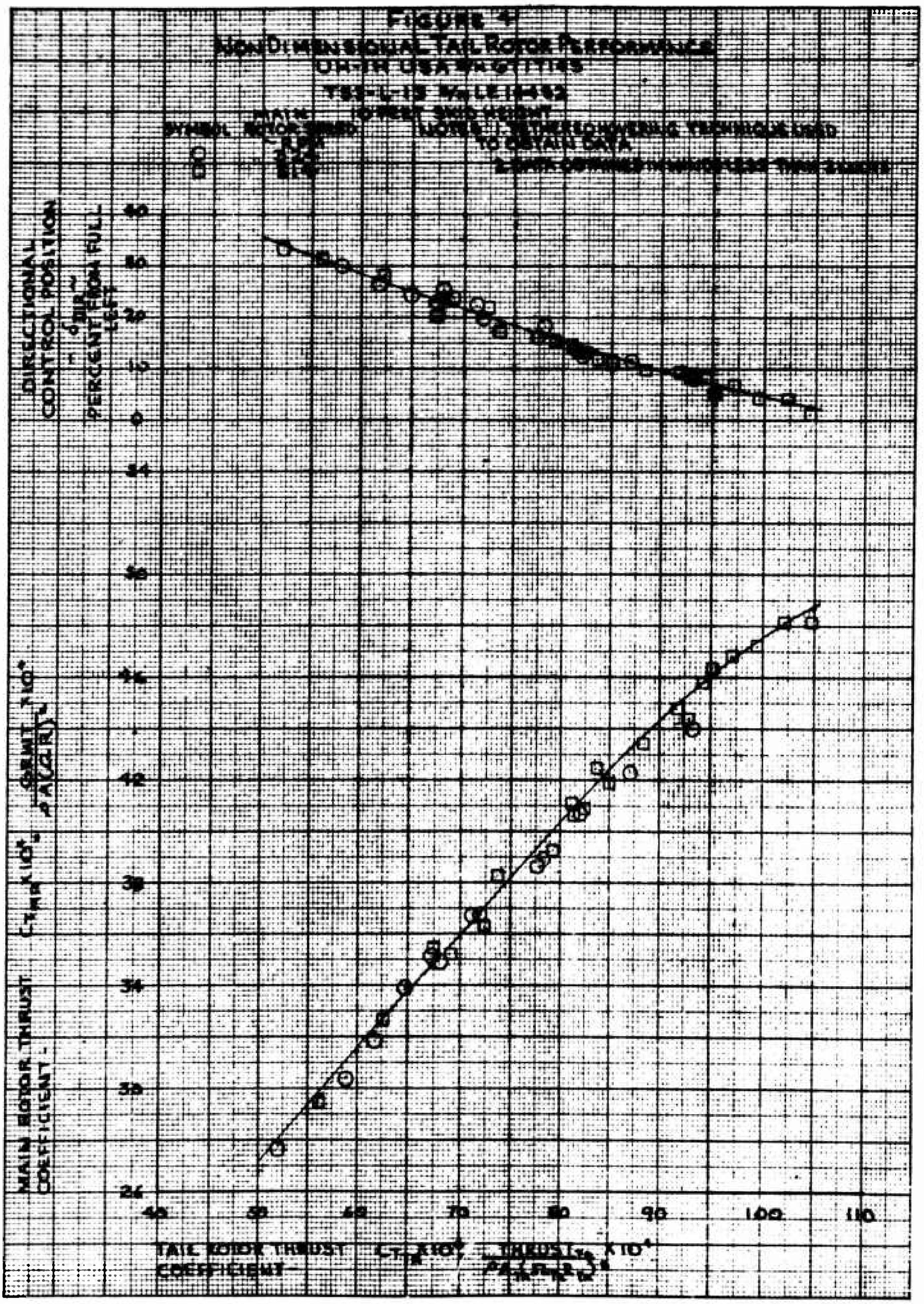


FIGURE 4
NONDIMENSIONAL TAIL ROTOR PERFORMANCE
DIRECTIONAL CONTROL POSITION
PERCENT FROM FULL LEFT
MAIN ROTOR THRUST COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
40
50
60
70
80
90
100
110

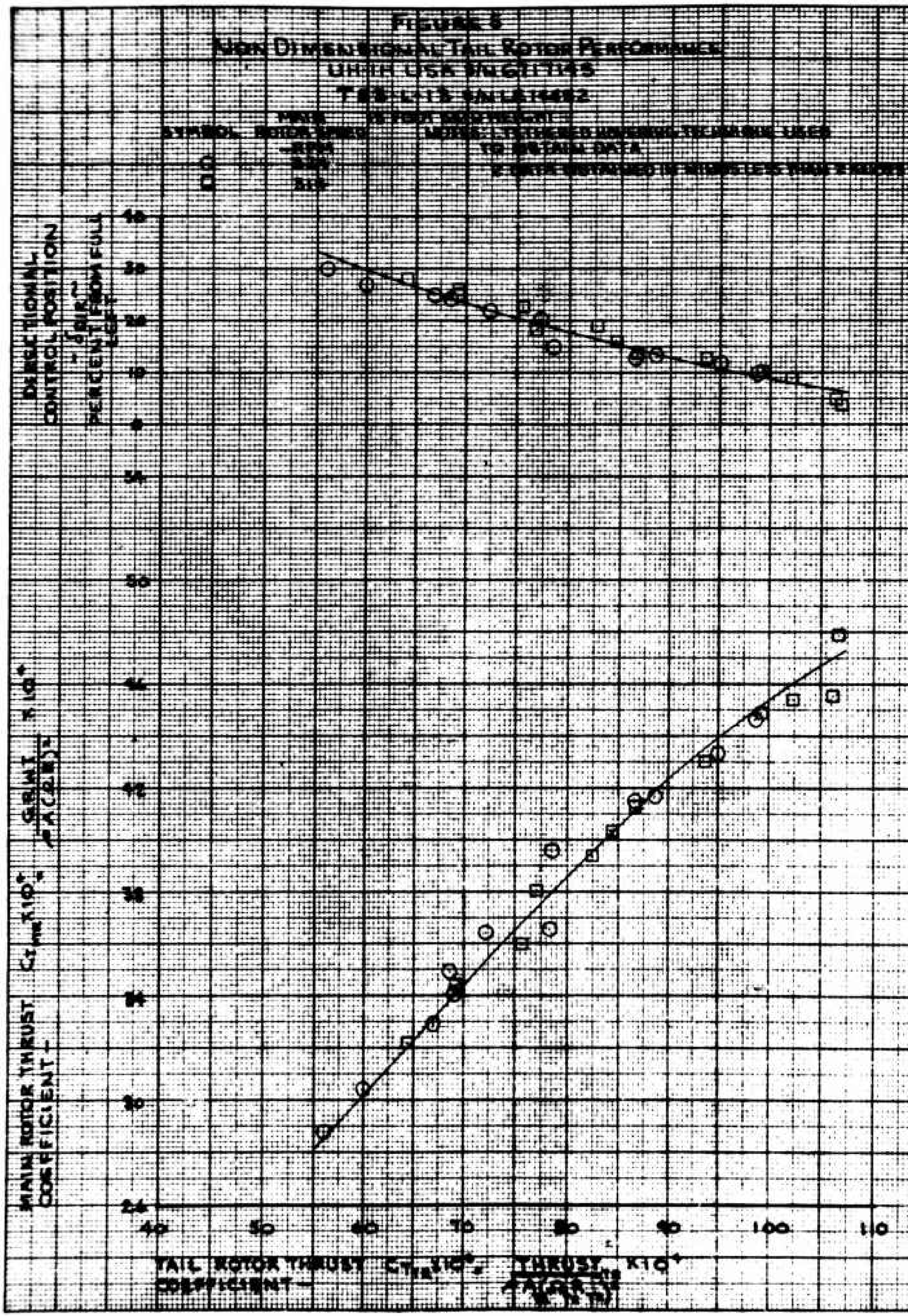


FIGURE 5
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
UH-1H USA SN 67-17145
T53-L-13 SN LE14452
DIRECTIONAL CONTROL POSITION
PERCENT FROM FULL LEFT
MAIN ROTOR THRUST COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
40
50
60
70
80
90
100
110

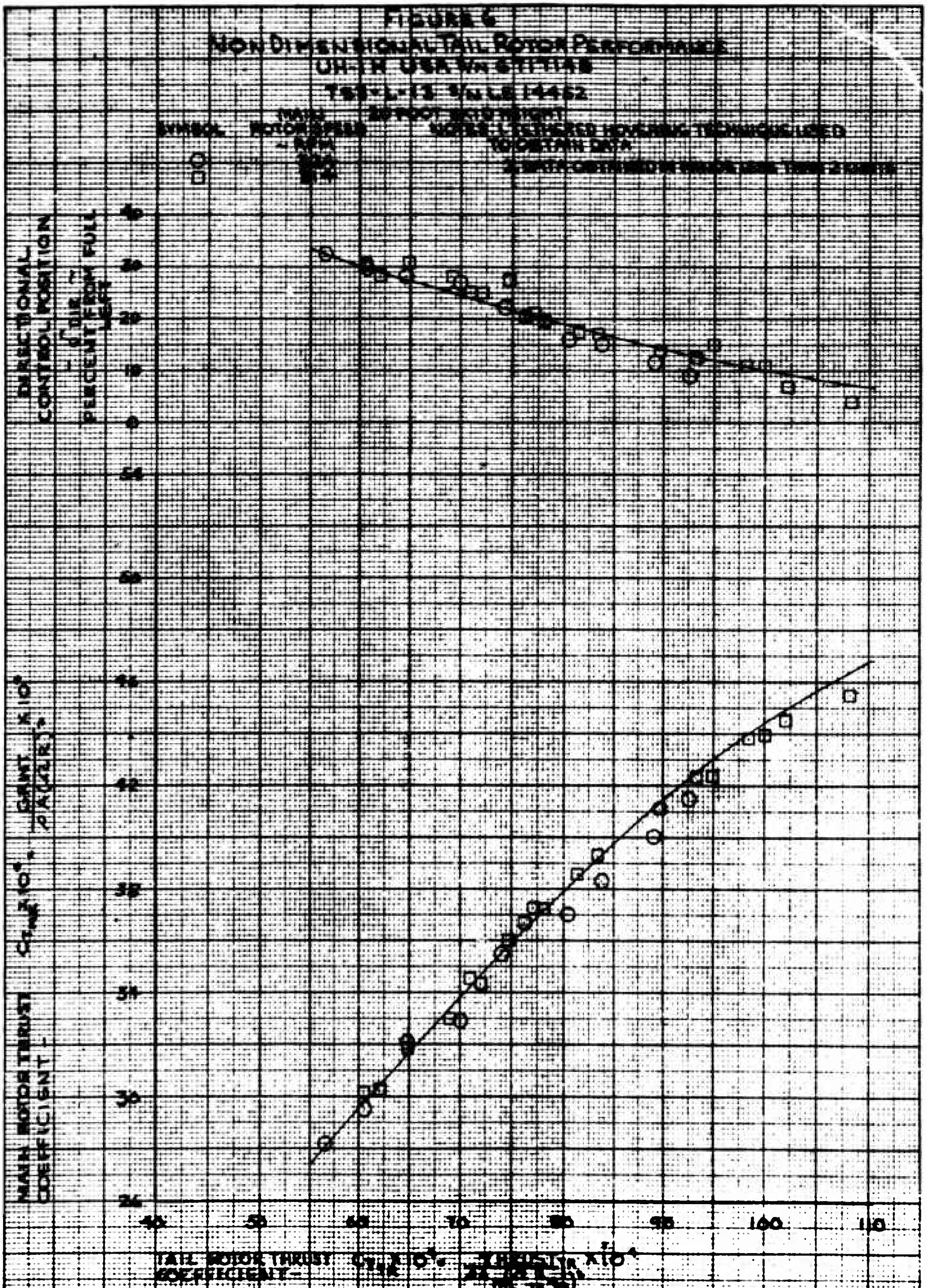


FIGURE 6
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
SYMBOL
DIRECTIONAL CONTROL POSITION
PERCENT FROM FULL LEFT
MAIN ROTOR THRUST COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
40
30
20
10
0
30
26
40
50
60
70
80
90
100
110

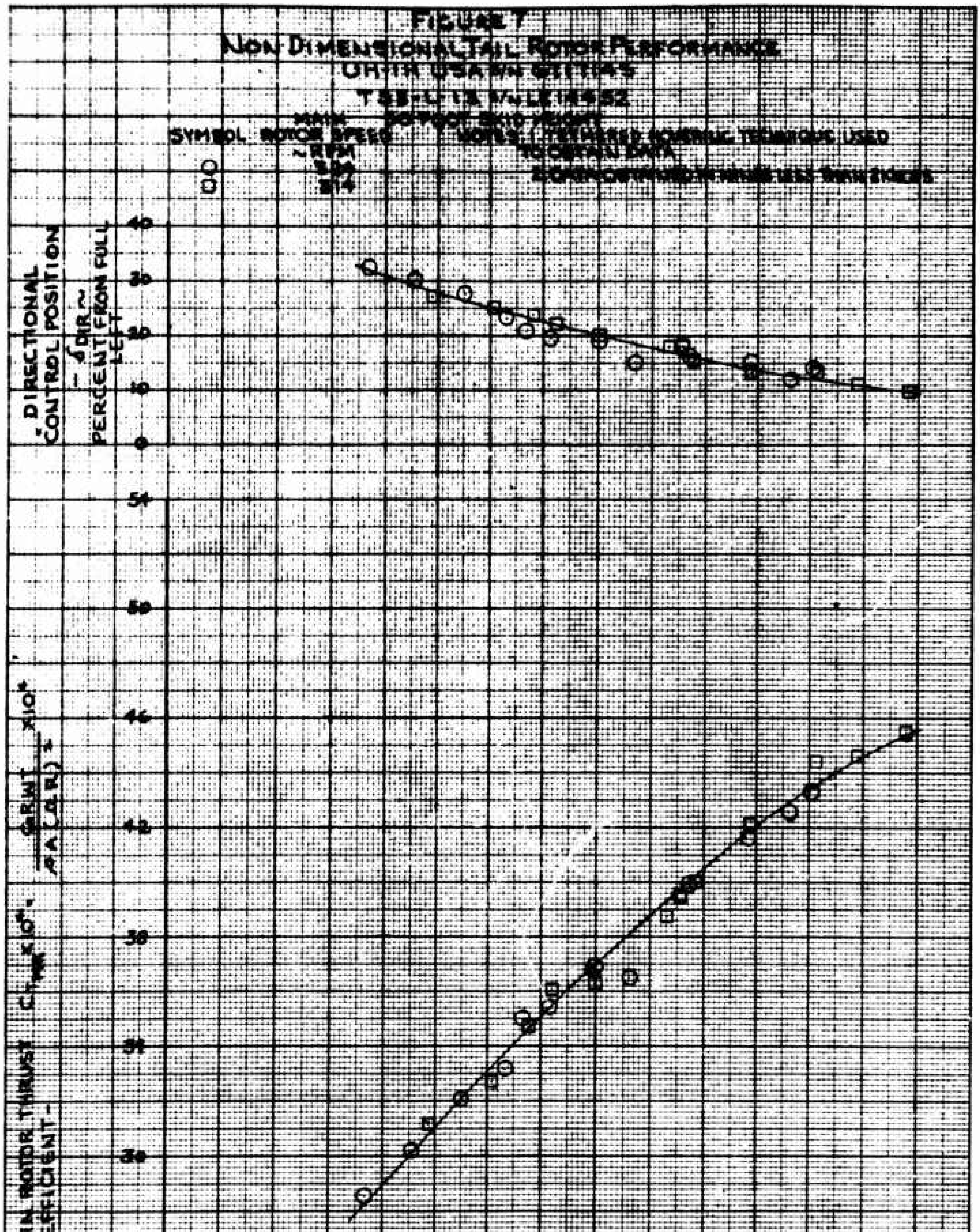


FIGURE 7
NON-DIMENSIONAL TAIL ROTOR PERFORMANCE
SYMBOL ROTOR SPEED
~RPM
DIRECTIONAL CONTROL POSITION
PERCENT FROM FULL LEFT
MAIN ROTOR THRUST COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
40
30
20
10
0
50
60
70
80
90
100
110

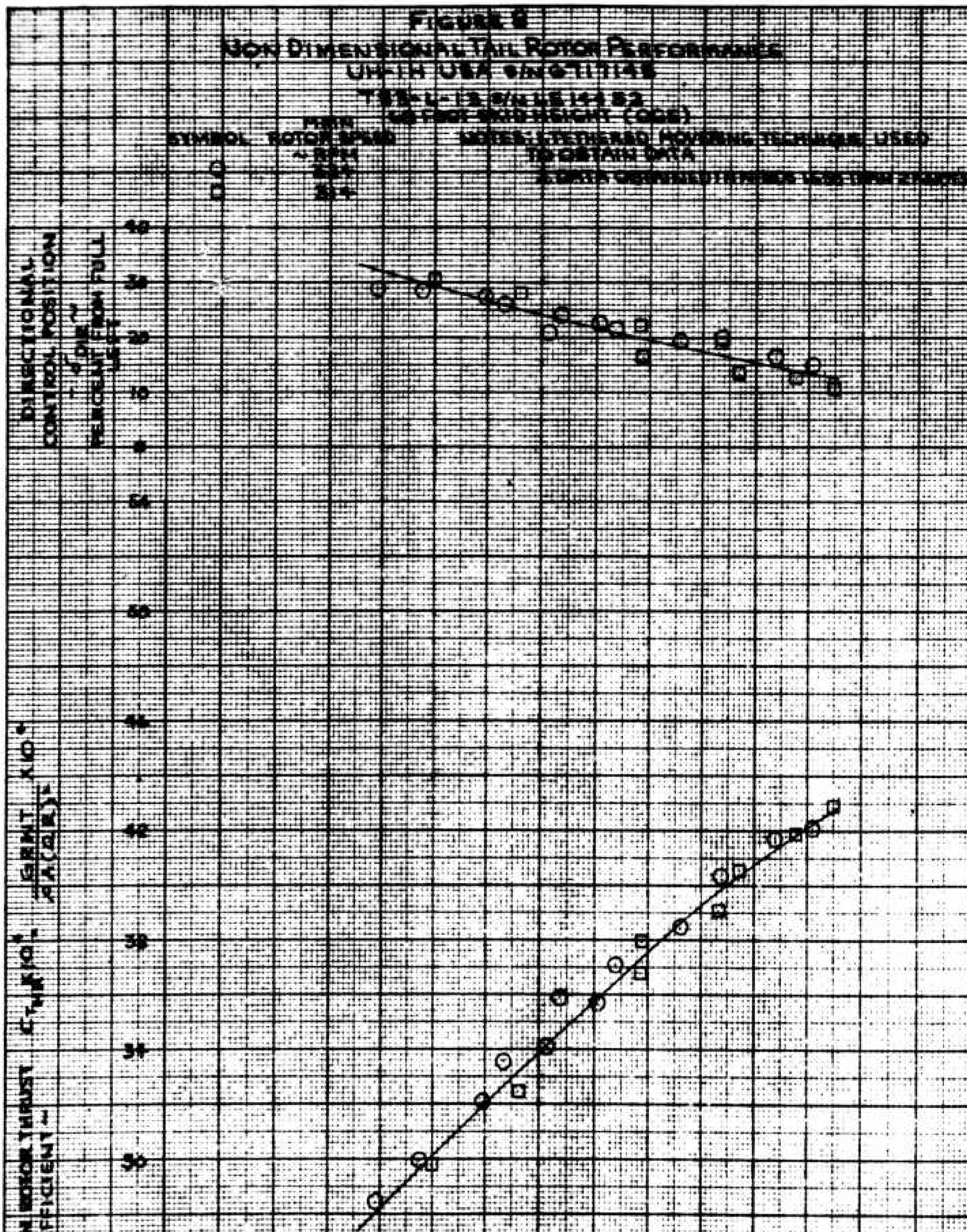


FIGURE 6
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
UH-1H USA S/N 67-17145
T53-L-13 S/N LE 14452
SYMBOL
ROTOR SPEED ~ RPM
NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
MAIN ROTOR THRUST COEFFICIENT ~
TAIL ROTOR THRUST COEFFICIENT ~
40
50
60
70
80
90
100
110

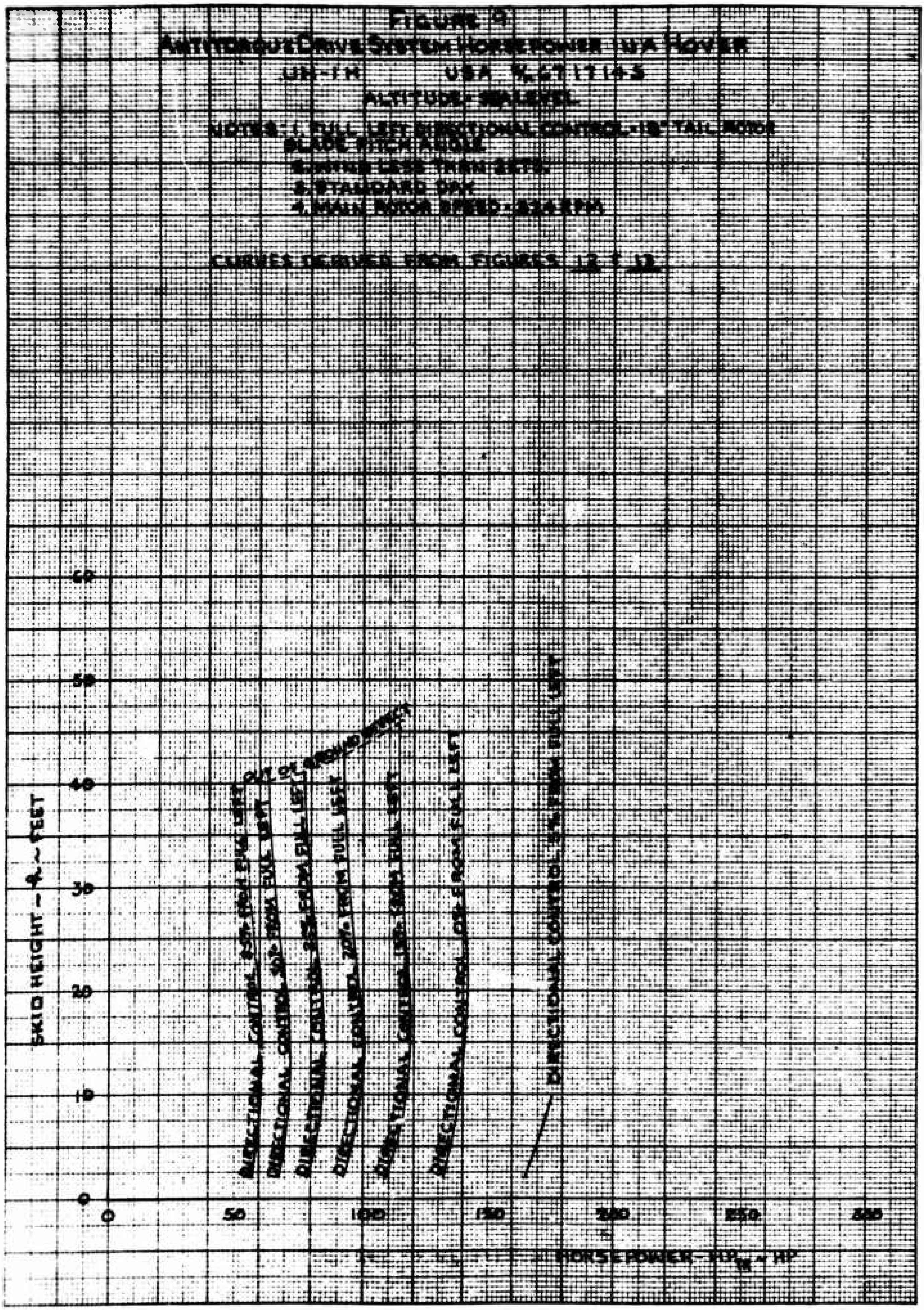
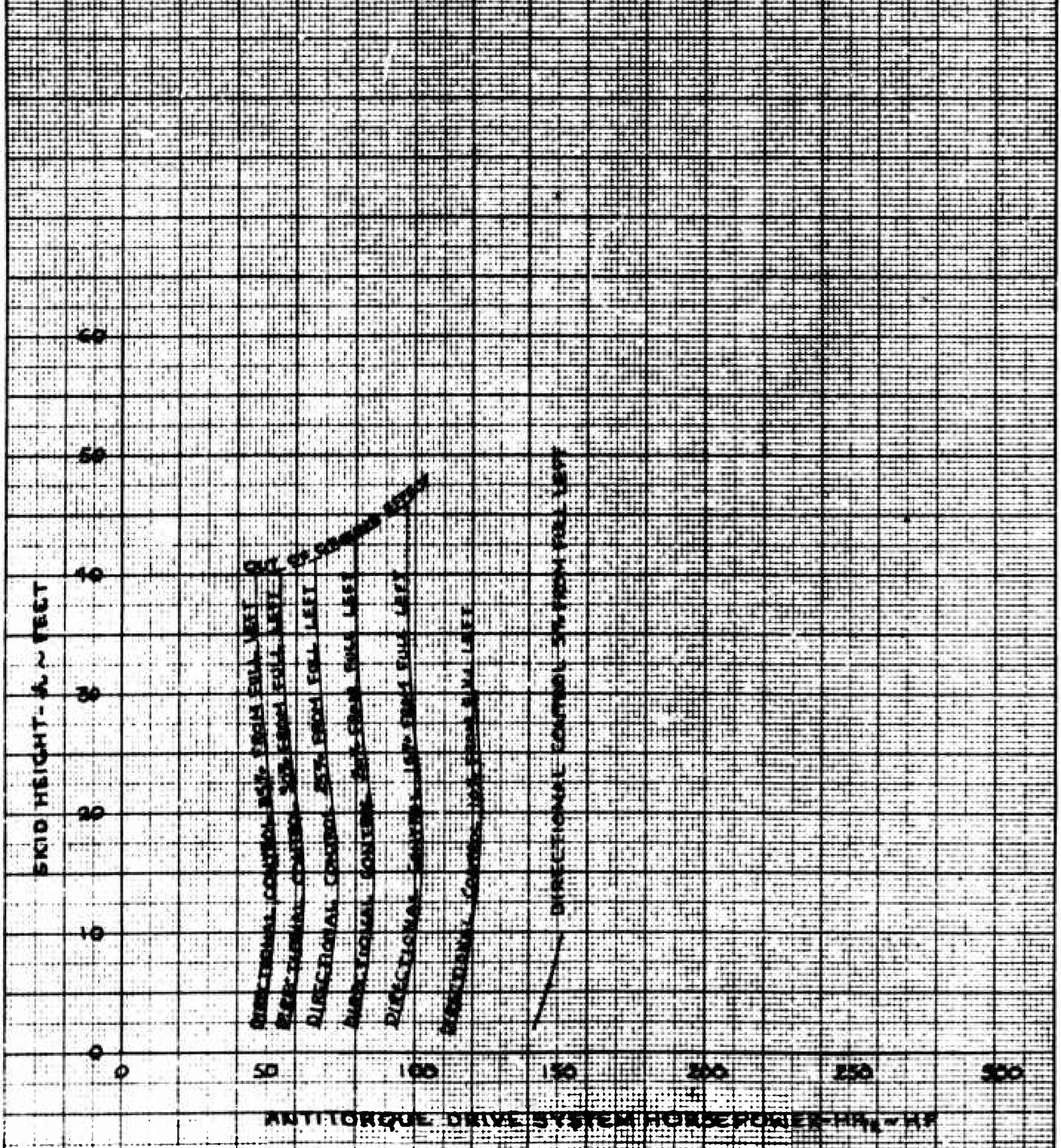

UH-1H
NOTES: 1. FULL LEFT DIRECTIONAL CONTROL
2. STANDARD DAY
CURVES DERIVED FROM FIGURES
OUT OF GROUND EFFECT
DIRECTIONAL CONTROL
SKID HEIGHT ~ FEET
HORSEPOWER ~ HP
0
10
20
30
40
50
60
0
50
100
150
200
250
300

FIGURE 10
ANTITORQUE DRIVE SYSTEM HORSEPOWER IN A HOVER
UH-1H USA S/N 6717148
ALTITUDE = 5000 FEET

NOTES: 1. FULL LEFT DIRECTIONAL CONTROL = 18° TAIL ROTOR BLADE PITCH ANGLE
2. WIND LESS THAN 3 KTS.
3. STANDARD DAY
4. MAIN ROTOR SPEED = 324 RPM

CURVES DERIVED FROM FIGURES 12 & 13



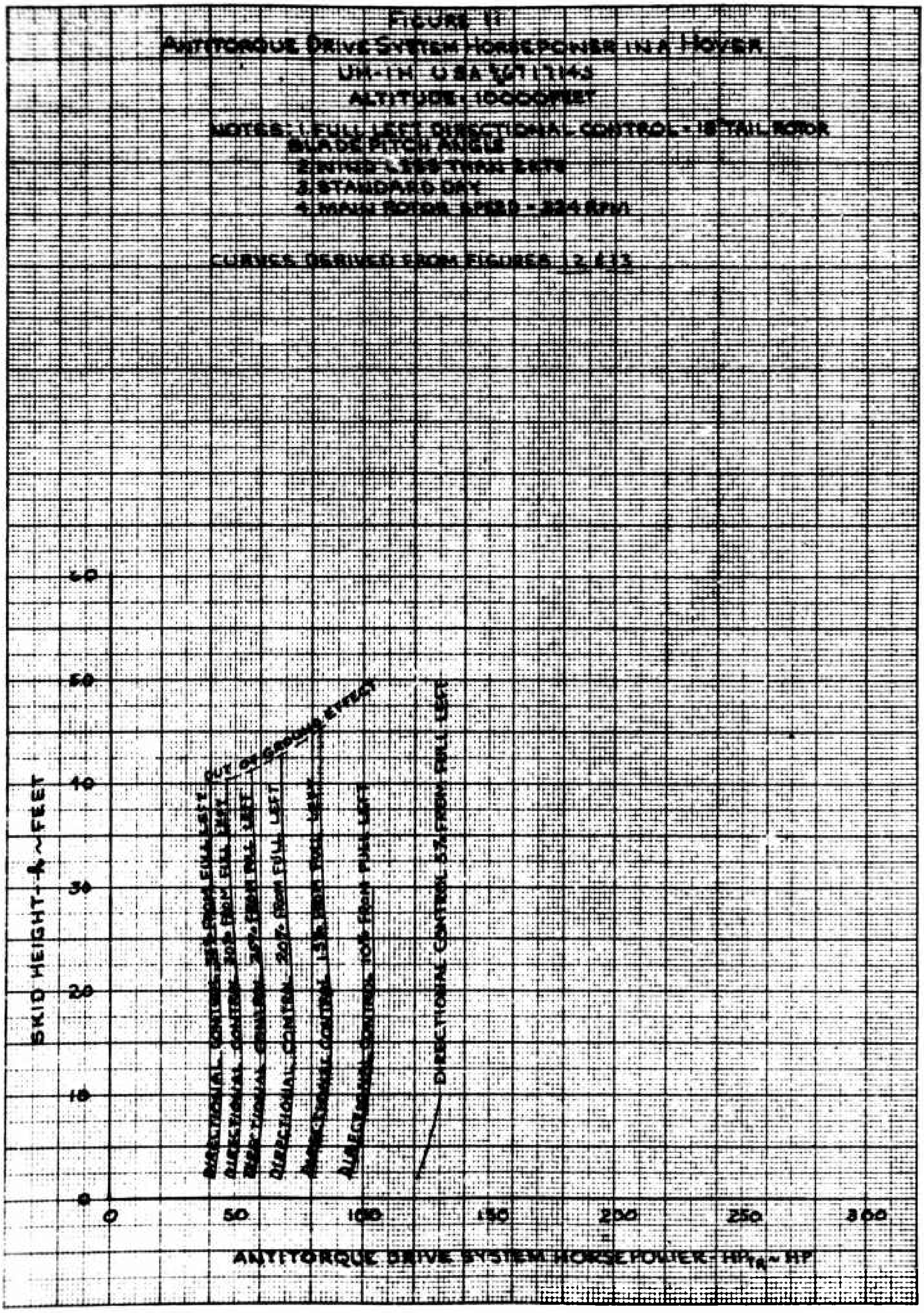


FIGURE 11
ANTITORQUE DRIVE SYSTEM HORSEPOWER IN A HOVER
UH-1H USA S/N
ALTITUDE: 10000 FEET
NOTES: 1. FULL LEFT DIRECTIONAL CONTROL = 18° TAIL ROTOR BLADE PITCH ANGLE
2. WIND LESS THAN 3 KTS
3. STANDARD DAY
4. MAIN ROTOR SPEED = 324 RPM
CURVES DERIVED FROM FIGURES 12 & 13
OUT OF GROUND EFFECT
DIRECTIONAL CONTROL 35% FROM FULL LEFT
DIRECTIONAL CONTROL 30% FROM FULL LEFT
DIRECTIONAL CONTROL 25% FROM FULL LEFT
DIRECTIONAL CONTROL 20% FROM FULL LEFT
DIRECTIONAL CONTROL 15% FROM FULL LEFT
DIRECTIONAL CONTROL 10% FROM FULL LEFT
DIRECTIONAL CONTROL 5% FROM FULL LEFT
SKID HEIGHT ~ FEET
0
10
20
30
40
50
60
0
50
100
150
200
250
300
ANTITORQUE DRIVE SYSTEM HORSEPOWER ~ HP

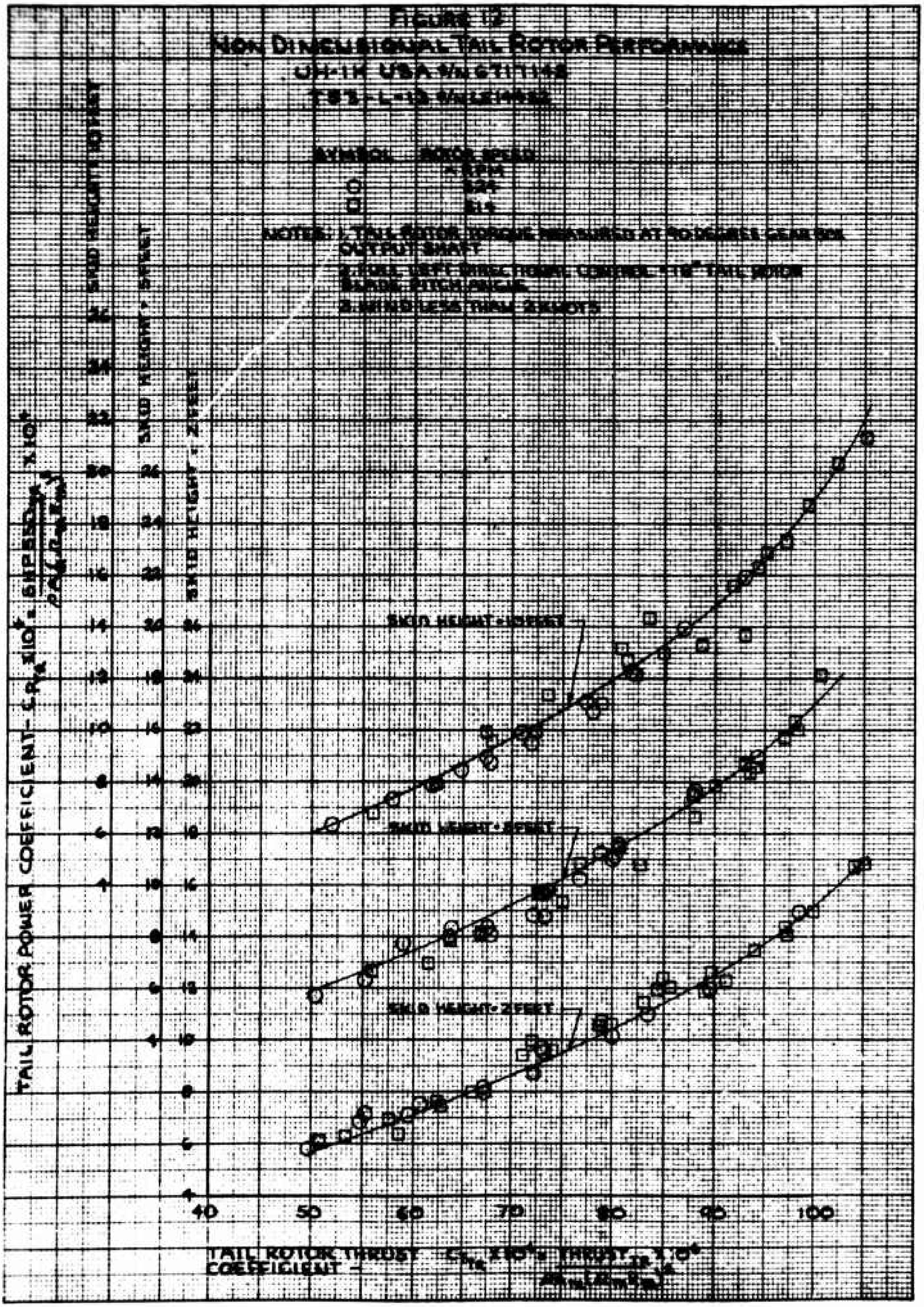


FIGURE 12
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
SYMBOL ROTOR SPEED ~RPM
NOTES: 1. TAIL ROTOR TORQUE MEASURED AT 90 DEGREE GEAR BOX OUTPUT SHAFT
3. WIND LESS THAN 2 KNOTS
SKID HEIGHT = 15 FEET
SKID HEIGHT = 5 FEET
SKID HEIGHT = 2 FEET
TAIL ROTOR POWER COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
40
50
60
70
80
90
100

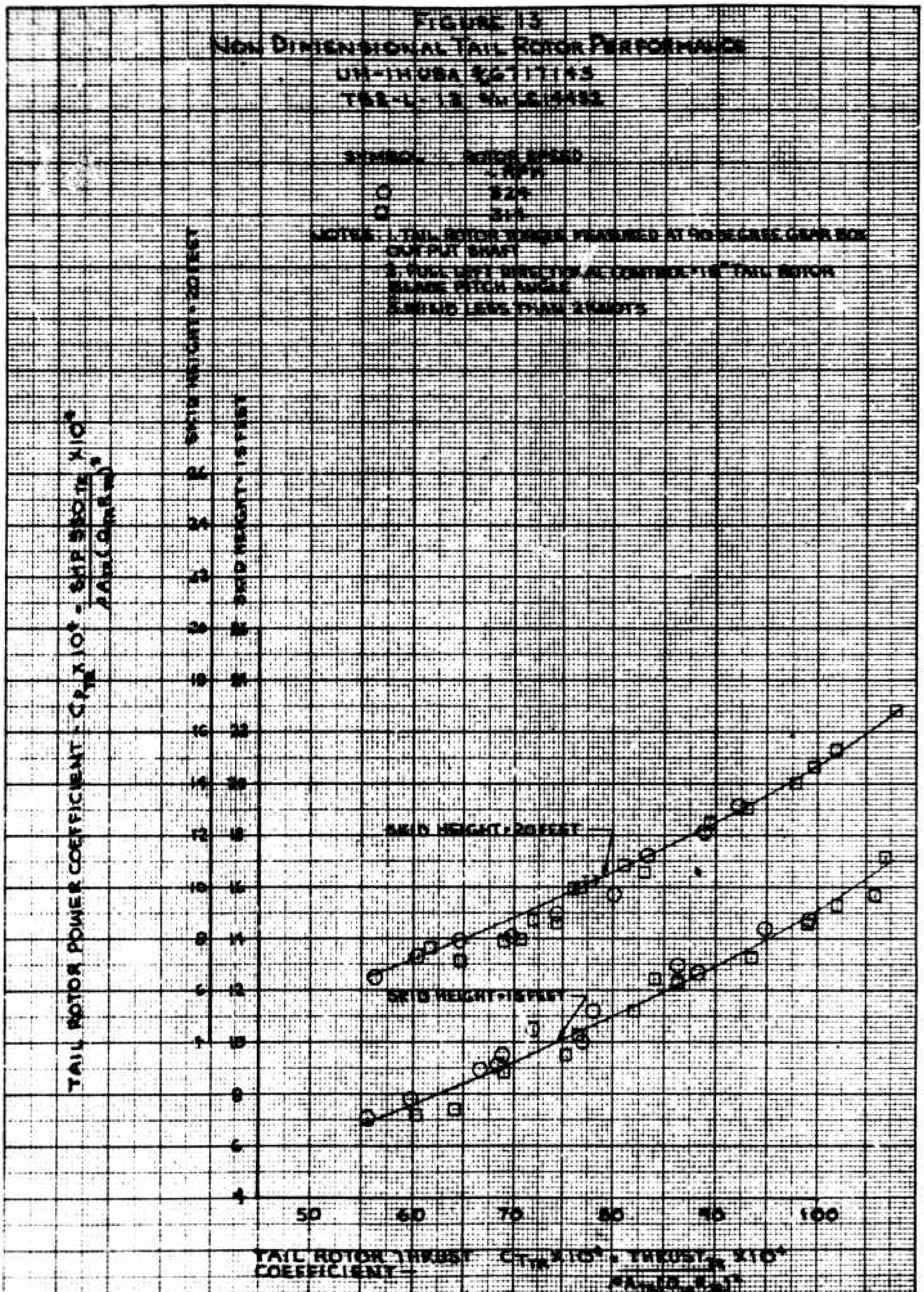


FIGURE 13
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
UH-1H USA S/N 67-17145
SYMBOL ROTOR SPEED ~ RPM
NOTES: 1. TAIL ROTOR TORQUE MEASURED AT 90 DEGREE GEAR BOX OUTPUT SHAFT
2. FULL LEFT DIRECTIONAL CONTROL
3. WIND LESS THAN 2 KNOTS
SKID HEIGHT = 20 FEET
SKID HEIGHT = 15 FEET
TAIL ROTOR POWER COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
50
60
70
80
90
100

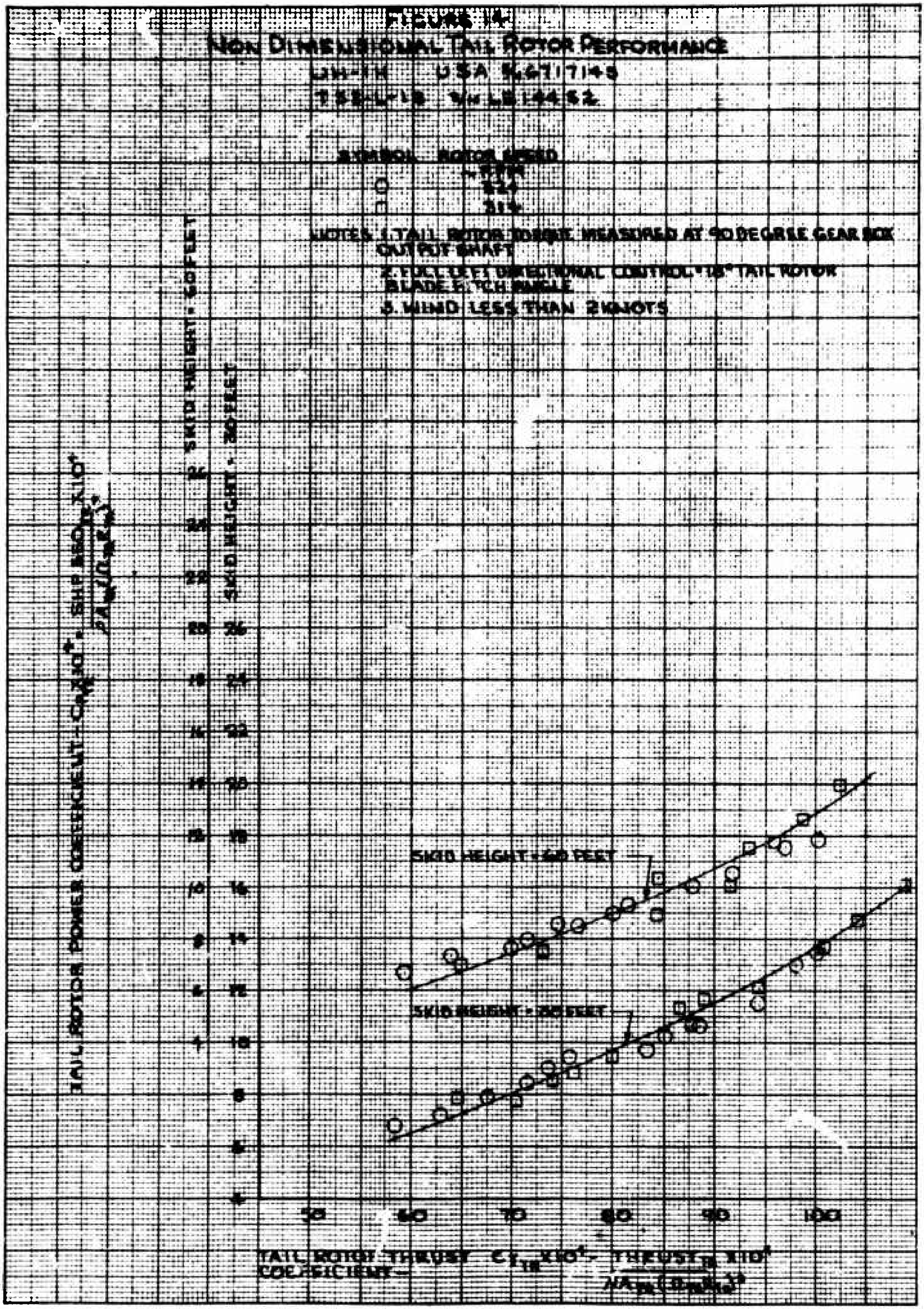
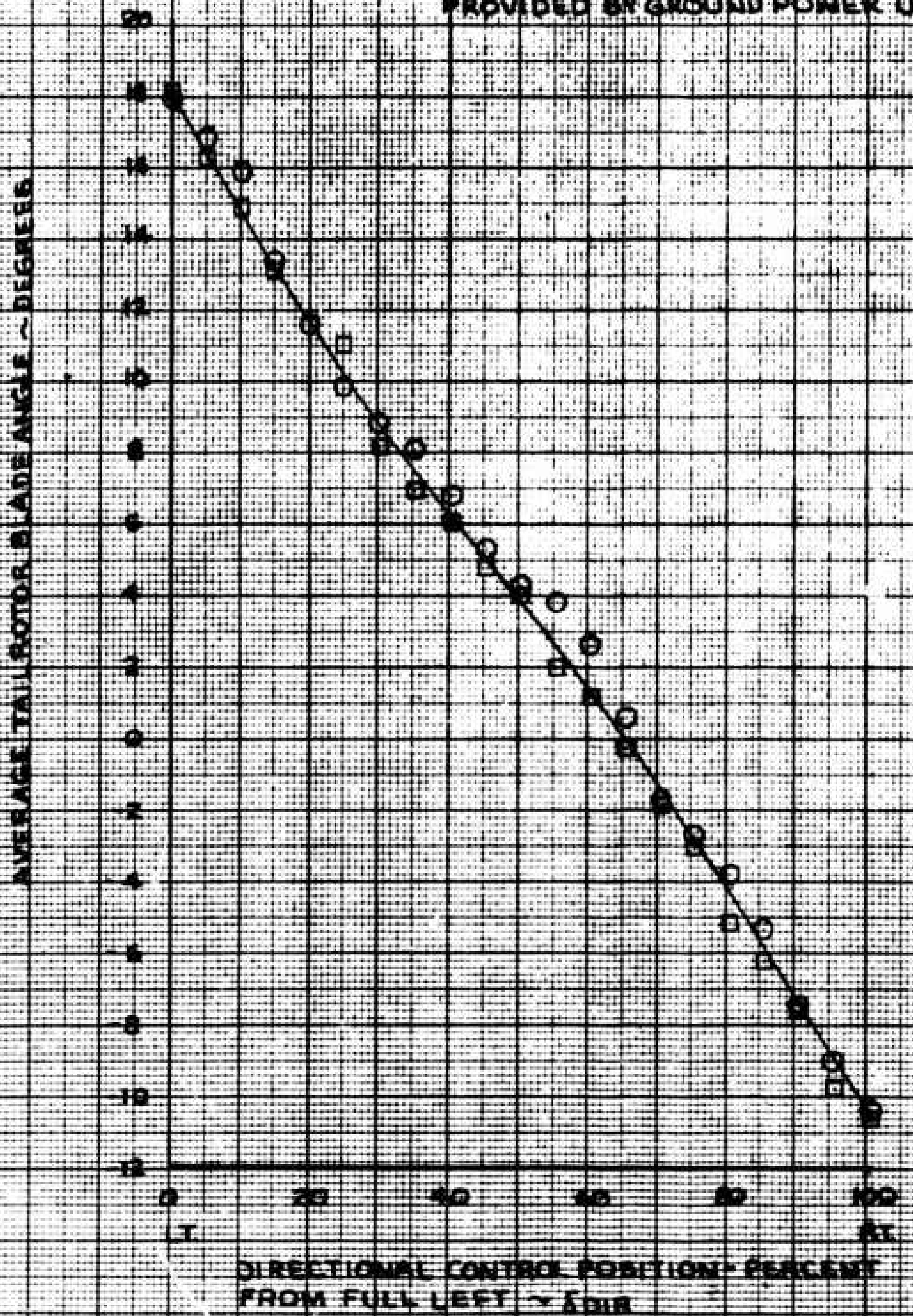


FIGURE 14
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
USA
SYMBOL ROTOR SPEED
~RPM
324
314
NOTES 1. TAIL ROTOR TORQUE MEASURED AT 90 DEGREE GEAR BOX OUTPUT SHAFT
2. FULL LEFT DIRECTIONAL CONTROL
BLADE PITCH ANGLE
3. WIND LESS THAN 2 KNOTS
SKID HEIGHT = 60 FEET
SKID HEIGHT = 20 FEET
TAIL ROTOR POWER COEFFICIENT
TAIL ROTOR THRUST COEFFICIENT
50
60
70
80
90
100

FIGURE 15
AVERAGE TAIL ROTOR BLADE ANGLE VS DIRECTIONAL CONTROL POSITION
UH-1H USA S/N 67-17145

○ ~ UP - ZERO TO 100 PERCENT DIRECTIONAL CONTROL DISPLACEMENT
□ ~ DOWN - 100 PERCENT TO ZERO DIRECTIONAL CONTROL DISPLACEMENT

NOTES: 1 AVERAGE TAIL ROTOR BLADE ANGLE IS EQUAL TO THE SUM OF THE WHITE AND RED BLADE ANGLES DIVIDED BY TWO
2 ROTOR STATIC
3 HYDRAULIC BOOST SYSTEM ON
4 HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS



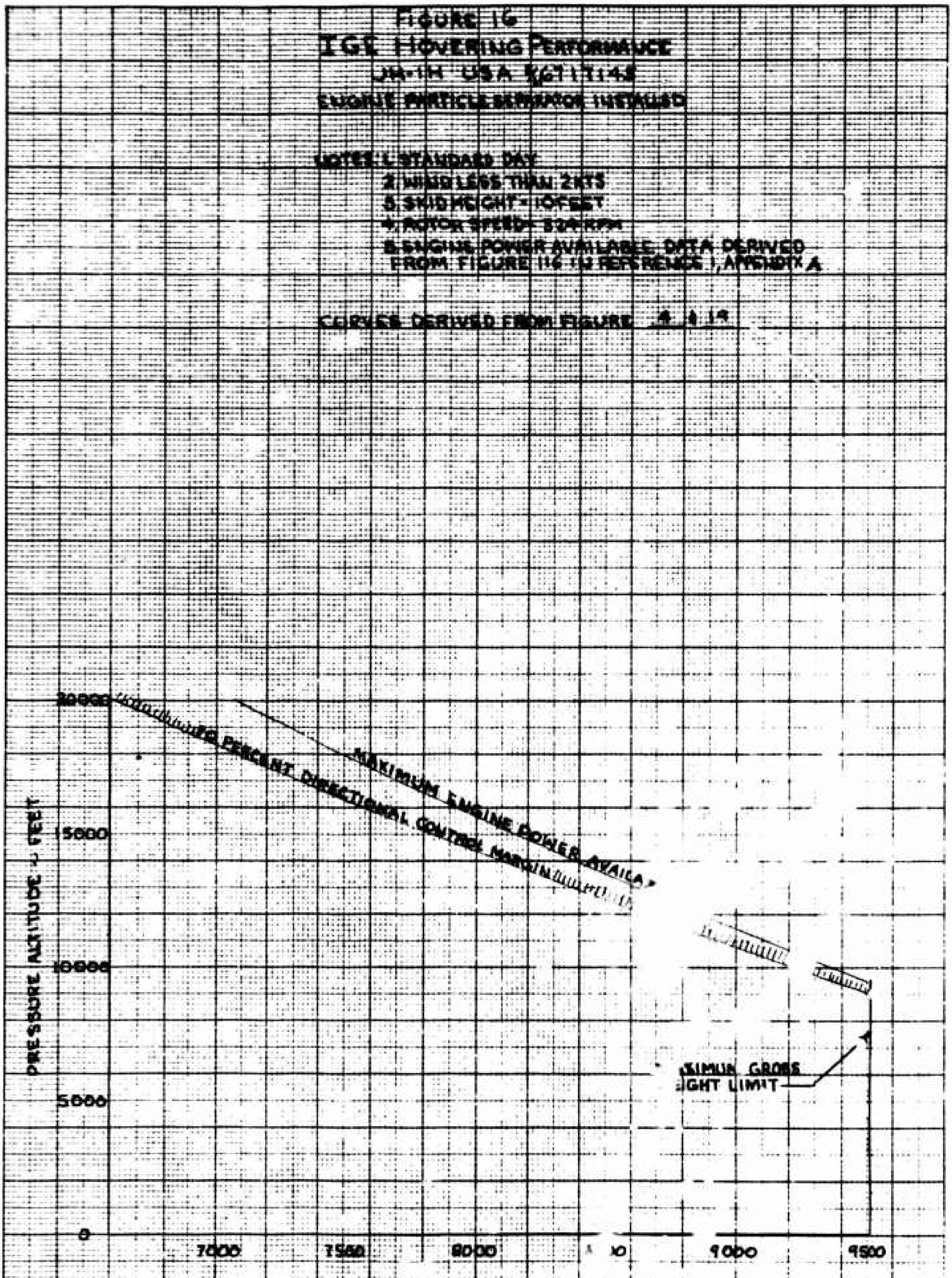
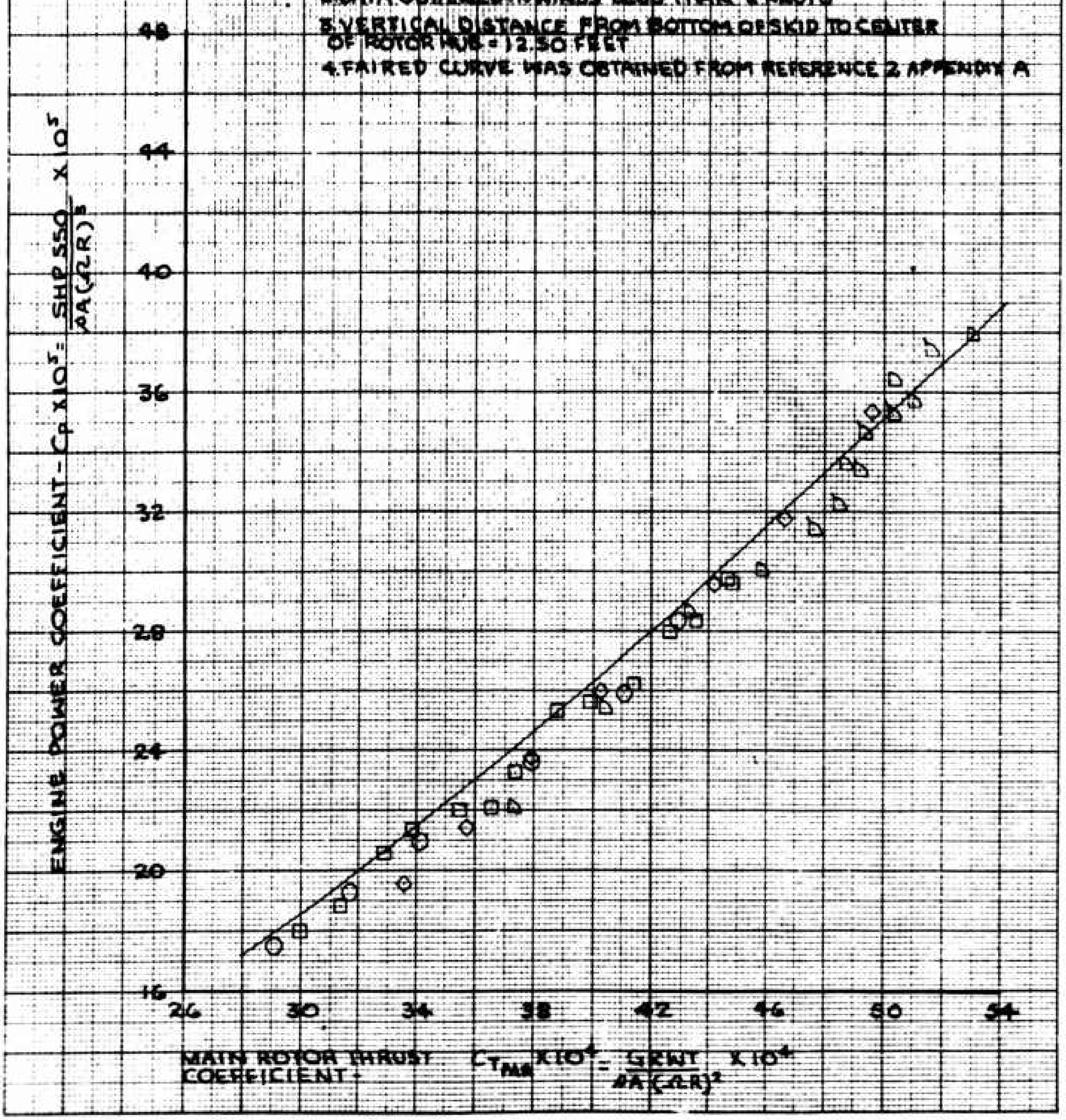


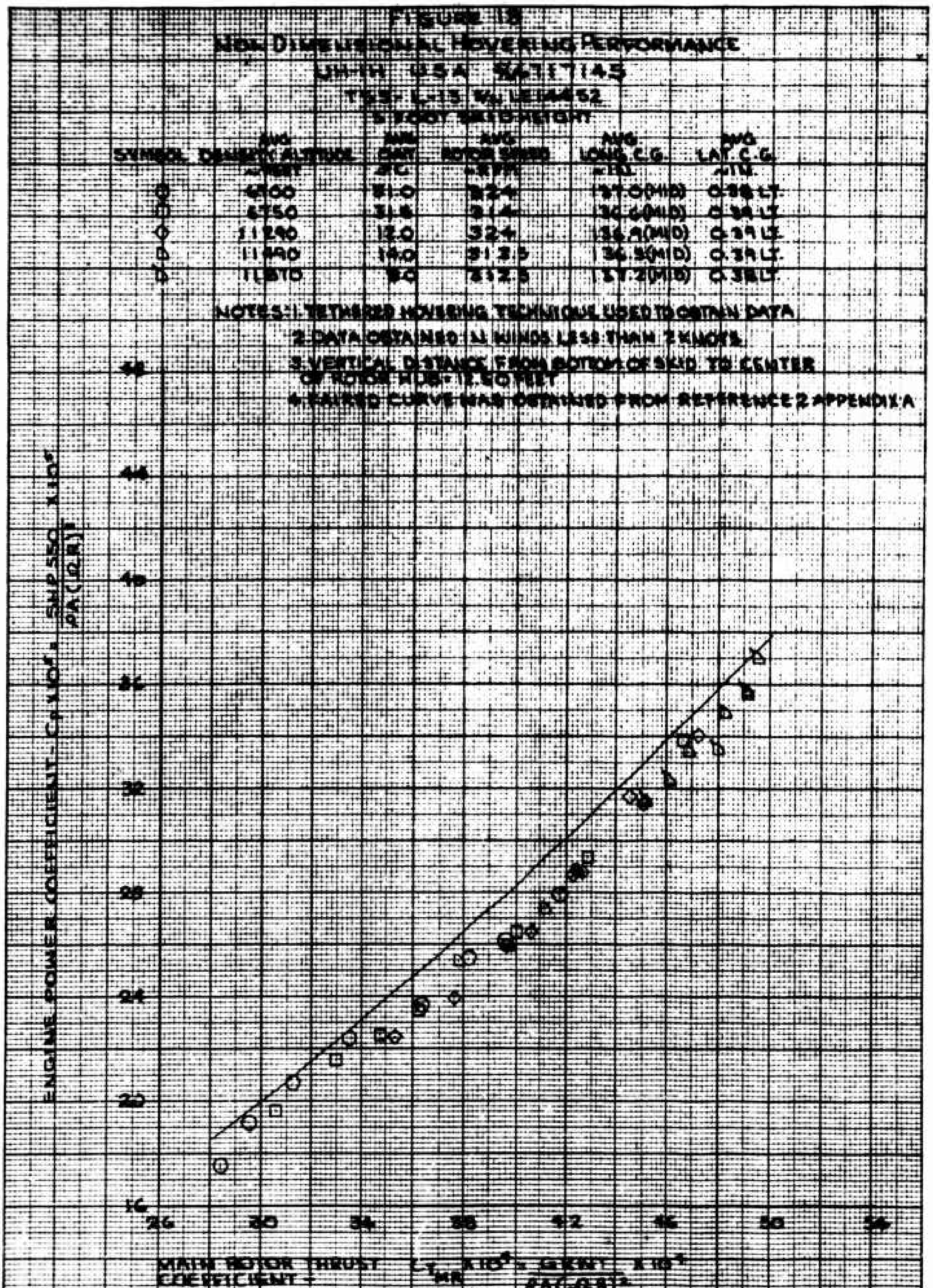
FIGURE 16
IGE HOVERING PERFORMANCE
NOTES: 1. STANDARD DAY
2. WIND LESS THAN 2 KTS
3. SKID HEIGHT = 10 FEET
CURVES DERIVED FROM FIGURE
20000
15000
10000
5000
0
7000
7500
8000
9000
9500
PRESSURE ALTITUDE ~ FEET
GROSS WEIGHT ~ POUND
MAXIMUM ENGINE POWER AVAILA
10 PERCENT DIRECTIONAL CONTROL MARGIN
GHT LIMIT

FIGURE 17
NON DIMENSIONAL HOVERING PERFORMANCE
UH-1H USA S/N 67-17145
T53-L-13 S/N LE 14452
2 FOOT SKID HEIGHT

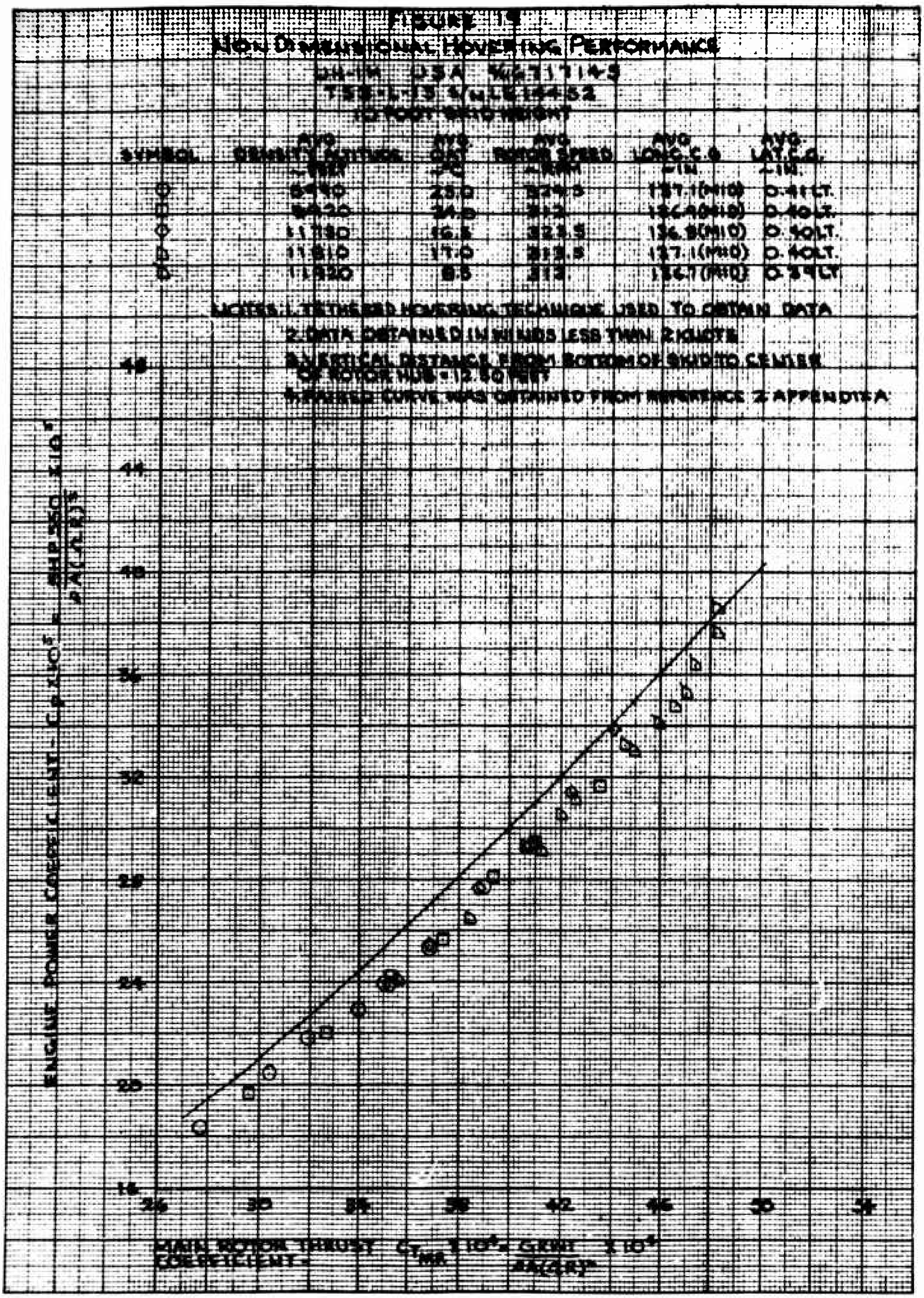
| SYMBOL | AVG. DENSITY ALTITUDE ~ FEET | AVG. OAT ~ °C | AVG. ROTOR SPEED ~ RPM | AVG. LONG. C.G. ~ IN. | AVG. LAT. C.G. ~ IN. |
|---|---|---|---|---|---|
| ○ | 6840 | 32.0 | 324 | 136.8(MID) | 0.40 LT. |
| □ | 5700 | 22.0 | 314 | 136.6(MID) | 0.39 LT. |
| ◇ | 11 320 | 12.5 | 324 | 137.8(MID) | 0.38 LT. |
| ◁ | 11 370 | 13.0 | 313 | 137.5(MID) | 0.38 LT. |
| ◁ | 10 820 | 7.5 | 313 | 137.5(MID) | 0.38 LT. |

NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. DATA OBTAINED IN WINDS LESS THAN 2 KNOTS
3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 12.50 FEET
4. FAIRED CURVE WAS OBTAINED FROM REFERENCE 2 APPENDIX A





NON DIMENSIONAL HOVERING PERFORMANCE
UH-1H USA
SYMBOL
DENSITY ALTITUDE
AVG ROTOR SPEED
AVG LONG. C.G.
AVG LAT. C.G.
11290
12.0
324
NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. DATA OBTAINED IN WINDS LESS THAN 2 KNOTS
3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB
4. FAIRED CURVE WAS OBTAINED FROM REFERENCE 2 APPENDIX A
ENGINE POWER COEFFICIENT
16
20
24
28
32
36
40
44
26
30
34
38
42
46
50
54
MAIN ROTOR THRUST COEFFICIENT


NON DIMENSIONAL HOVERING PERFORMANCE
SYMBOL
AVG DENSITY ALTITUDE ~FEET
AVG OAT ~°C
AVG ROTOR SPEED ~RPM
AVG LONG. C.G. ~IN.
AVG LAT. C.G. ~IN.
11,130 16.5 323.5 136.9(MID) 0.40LT.
11,810 17.0 313.5 137.1(MID) 0.40LT.
NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. DATA OBTAINED IN WINDS LESS THAN 3 KNOTS
ENGINE POWER COEFFICIENT
MAIN ROTOR THRUST COEFFICIENT

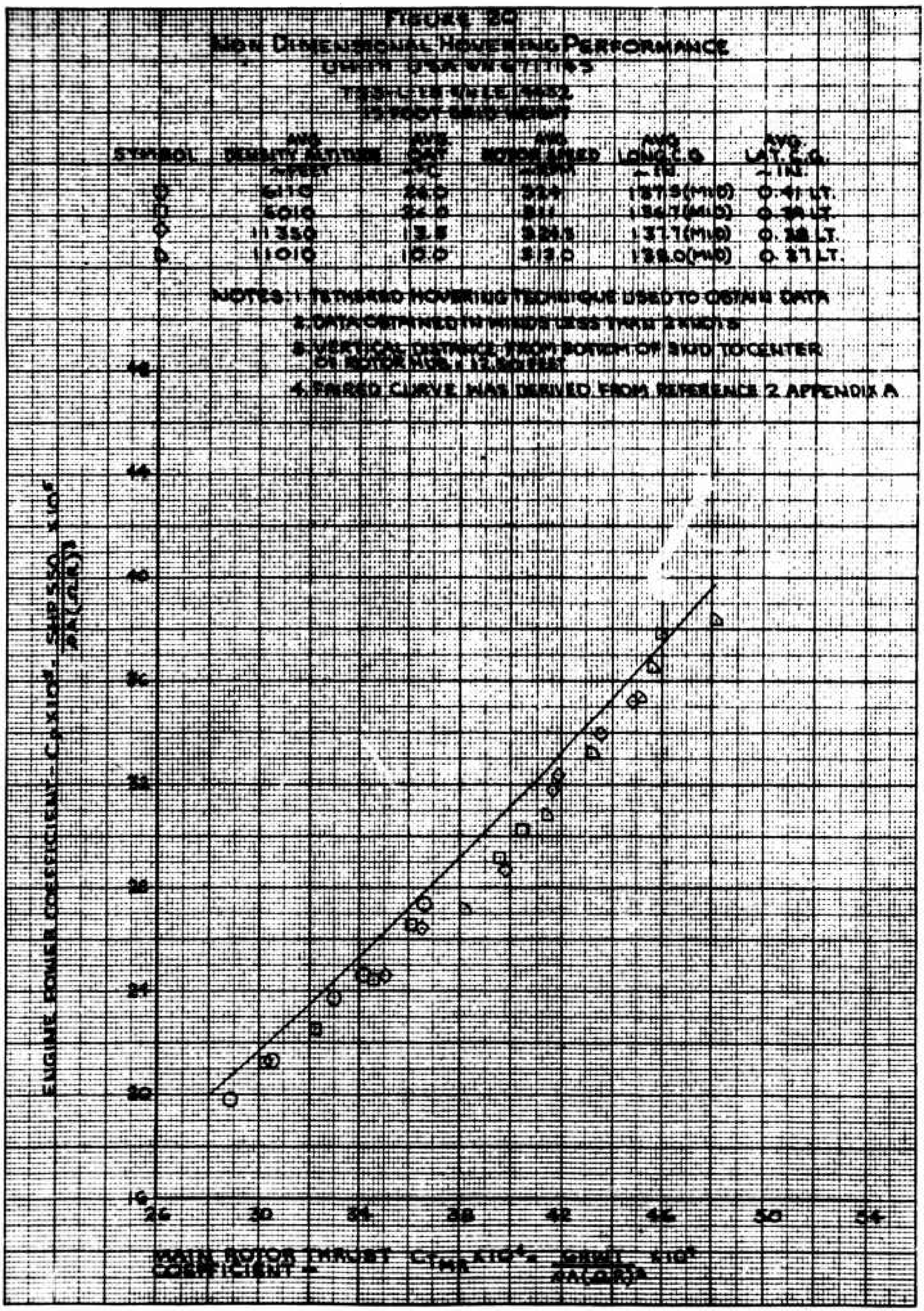

NON DIMENSIONAL HOVERING PERFORMANCE
SYMBOL
AVG.
DENSITY ALTITUDE
LONG. C.G.
LAT. C.G.
~IN.
6110
6010
11350
11010
13.5
10.0
137.7(MID)
NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER
4. FAIRED CURVE WAS DERIVED FROM REFERENCE 2 APPENDIX A
ENGINE POWER COEFFICIENT
MAIN ROTOR THRUST COEFFICIENT
16
20
24
28
32
36
40
44
48
26
30
34
38
42
46
50
54

FIGURE 2

NON DIMENSIONAL HOVERING PERFORMANCE

UH-1H USA S/N 67-17145

T53-L-13 S/N LE18452

[illegible]

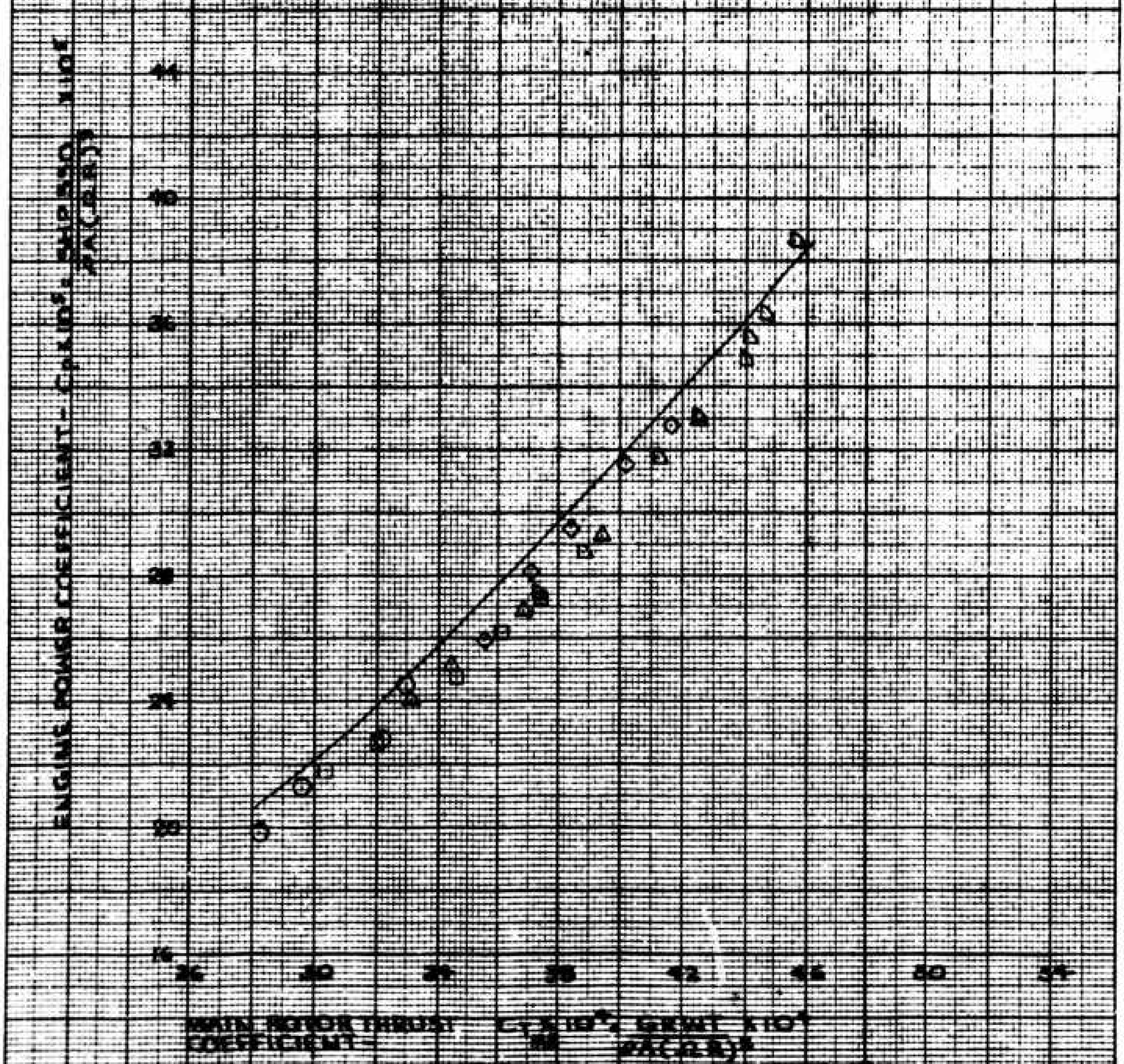
| SYMBOL | AVG. DENSITY ALTITUDE ~FEET | AVG. OAT ~°C | AVG. ROTOR SPEED ~RPM | AVG. LONG. C.G. ~IN. | AVG. LAT. C.G. ~IN. |
|---|---|---|---|---|---|
| ○ | 6180 | 27.8 | 324 | 136.8(M10) | 0.40 LT. |
| □ | 6040 | [illegible] | [illegible] | [illegible] | 0.39 LT. |
| ◇ | 6580 | 20.5 | 322 | 137.2(M10) | 0.38 LT. |
| △ | 11310 | 12.5 | 323.5 | 137.4(M10) | 0.38 LT. |
| ⌂ | 12130 | 20.0 | 323.5 | 136.9(M10) | 0.39 LT. |

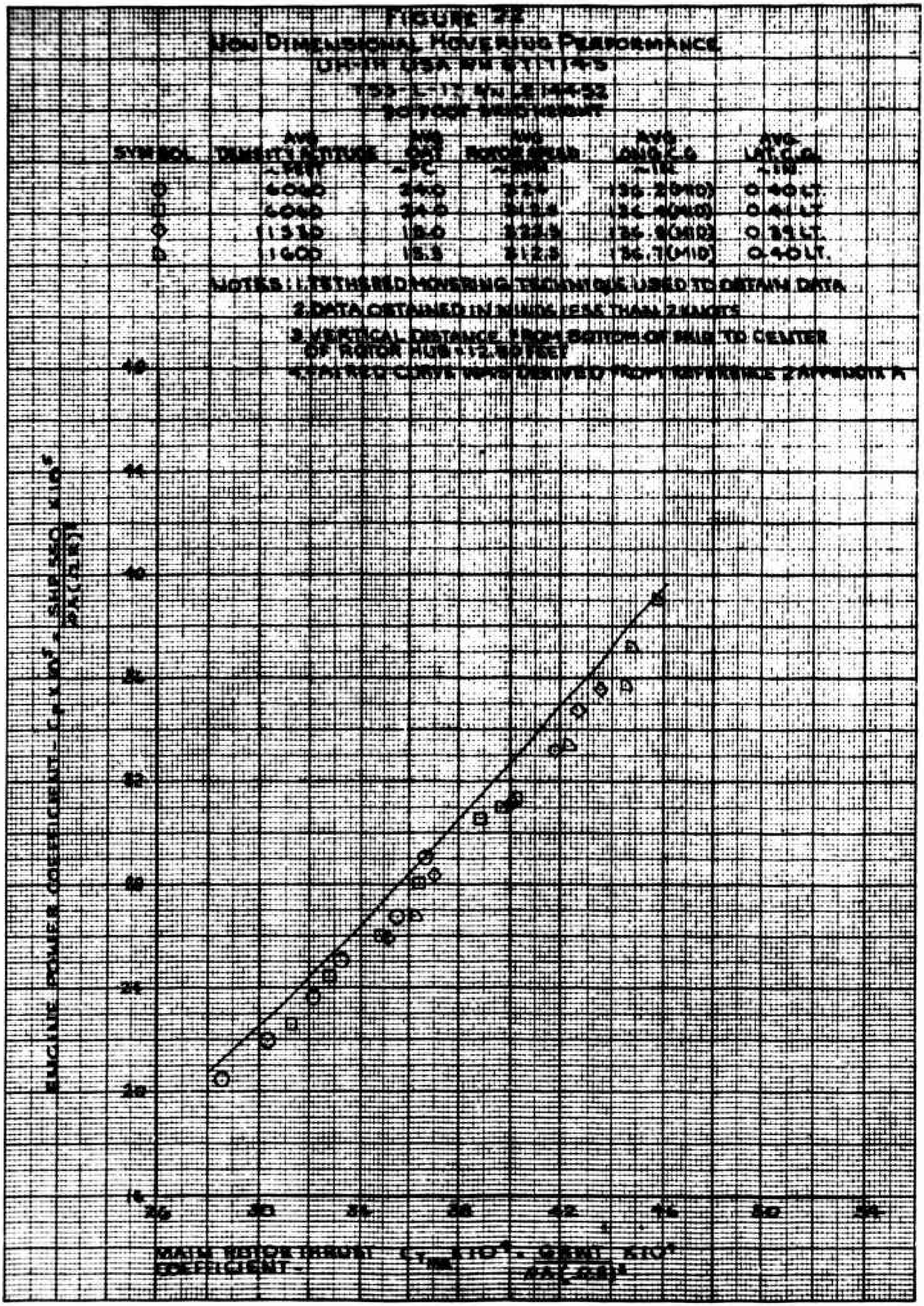
NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA

2. DATA OBTAINED IN WINDS LESS THAN 3 KNOTS

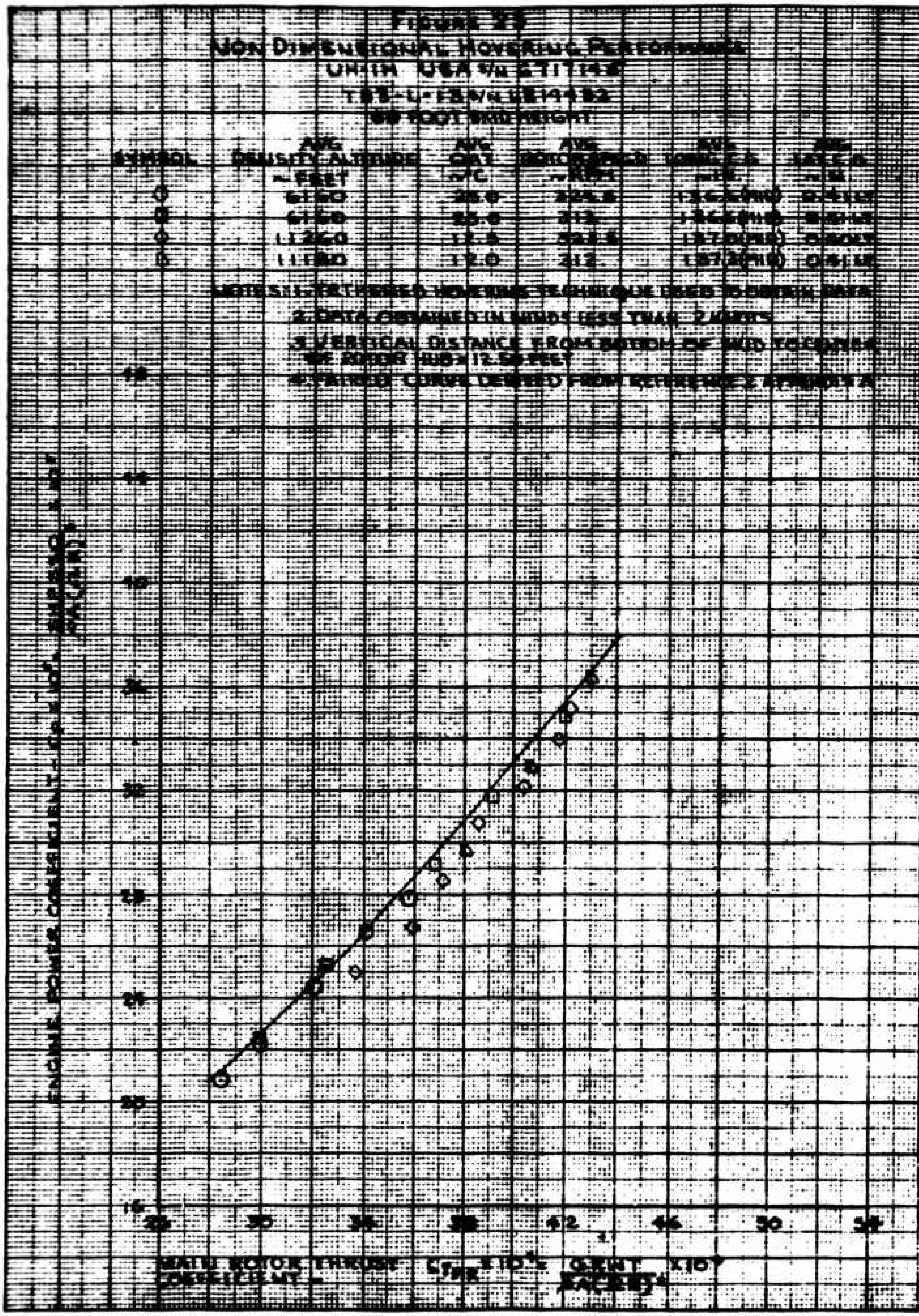
3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 12.50 FEET

4. FAIRED CURVE WAS OBTAINED FROM REFERENCE 2 APPENDIX A





NON DIMENSIONAL HOVERING PERFORMANCE
SYMBOL
AVG
1600
18.0
15.3
212.5
0.40 LT.
NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. DATA OBTAINED IN WINDS LESS THAN 2 KNOTS
3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB
ENGINE POWER COEFFICIENT
MAIN ROTOR THRUST COEFFICIENT


NON DIMENSIONAL HOVERING PERFORMANCE
SYMBOL
~FEET
6160
25.0
6160
25.0
11260
12.5
11180
12.0
212
2. DATA OBTAINED IN WINDS LESS THAN 2 KNOTS
3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 12.50 FEET
48
44
40
36
32
28
24
20
16
26
30
34
38
42
46
50
54
MAIN ROTOR THRUST COEFFICIENT

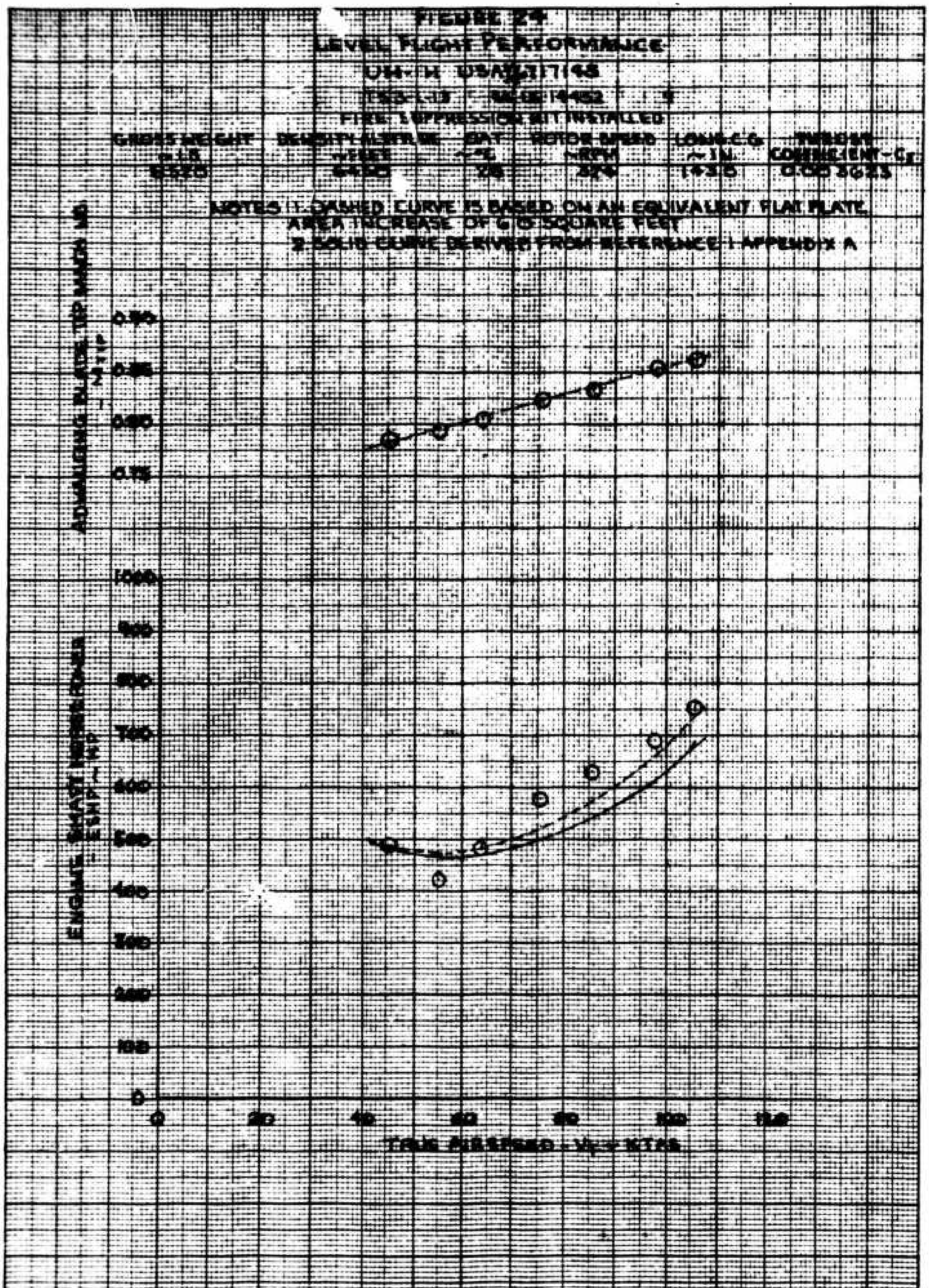
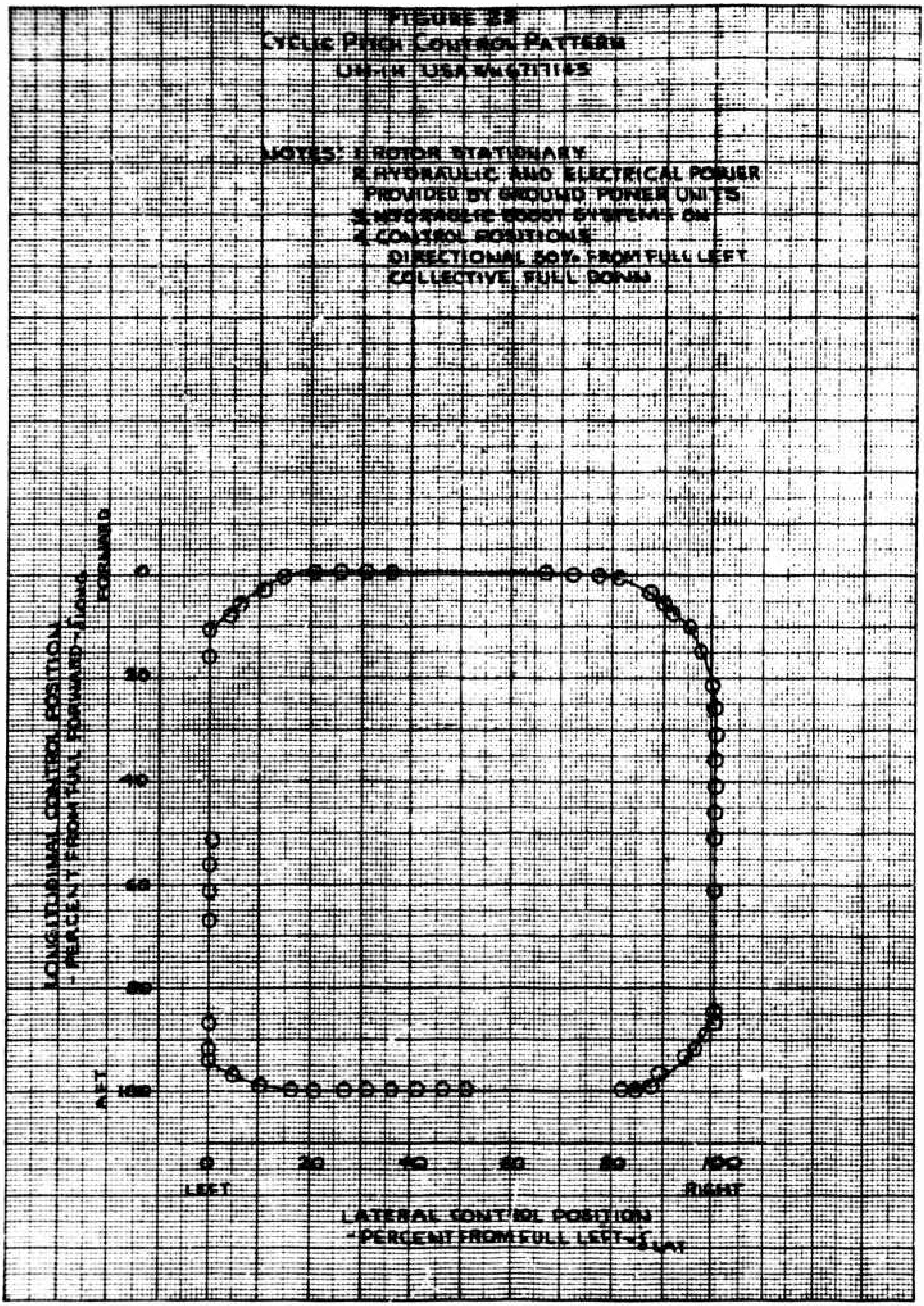


FIGURE 24
LEVEL FLIGHT PERFORMANCE
FIRE SUPPRESSION KIT INSTALLED
GROSS WEIGHT ~LB
DENSITY ALTITUDE ~FEET
OAT ~°C
ROTOR SPEED ~RPM
LONG. C.G. ~IN.
THRUST COEFFICIENT ~ $C_T$
NOTES 1. DASHED CURVE IS BASED ON AN EQUIVALENT FLAT PLATE
AREA INCREASE OF
2. SOLID CURVE DERIVED FROM REFERENCE 1 APPENDIX A
ADVANCING BLADE TIP MACH NO.
0.90
0.85
0.80
0.75
ENGINE SHAFT HORSEPOWER
1000
900
800
700
600
500
400
300
200
100
0
0
20
40
60
80
100
120
TRUE AIRSPEED ~ KTAS


CYCLIC PITCH CONTROL PATTERN
NOTES: 1. ROTOR STATIONARY
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS
4. CONTROL POSITIONS
DIRECTIONAL 50% FROM FULL LEFT
COLLECTIVE FULL DOWN
LONGITUDINAL CONTROL POSITION
- PERCENT FROM FULL FORWARD
FORWARD
AFT
0
20
40
60
80
100
LEFT
RIGHT
LATERAL CONTROL POSITION
- PERCENT FROM FULL LEFT

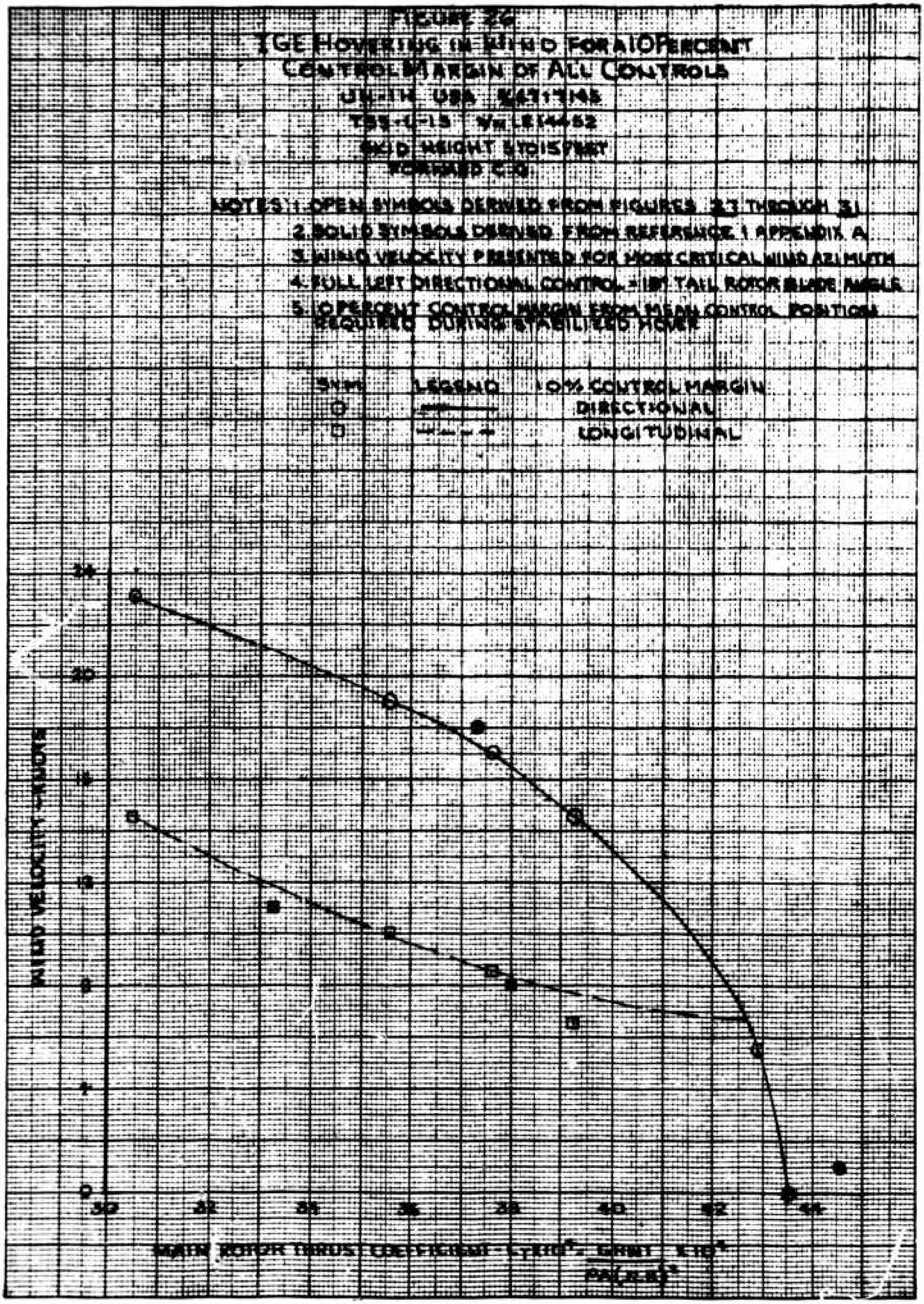


FIGURE 26
IGE HOVERING IN WIND FOR A 10 PERCENT
CONTROL MARGIN OF ALL CONTROLS
FORWARD C.G.
NOTES: 1. OPEN SYMBOLS DERIVED FROM FIGURES 27 THROUGH 31
2. SOLID SYMBOLS DERIVED FROM REFERENCE 1 APPENDIX A
3. WIND VELOCITY PRESENTED FOR MOST CRITICAL WIND AZIMUTH
4. FULL LEFT DIRECTIONAL CONTROL
5. 10 PERCENT CONTROL MARGIN FROM MEAN CONTROL POSITIONS REQUIRED DURING STABILIZED HOVER
SYM
LEGEND
10% CONTROL MARGIN
DIRECTIONAL
LONGITUDINAL
WIND VELOCITY ~ KNOTS
24
20
16
12
8
4
0
30
32
34
36
38
40
42
44
MAIN ROTOR THRUST COEFFICIENT

FIGURE 27
TRANSLATIONAL FLIGHT CONTROL MARGIN SUMMARY
UH-1H USA S/N 6717145

| AVG. DENSITY ALTITUDE ~FEET | AVG GRWT ~LB | AVG. LONG.C.G. ~IN. | AVG. LAT.C.G. ~IN. | AVG. ROTOR SPEED ~RPM | AVG. THRUST COEFF. $-C_T$ |
|---|---|---|---|---|---|
| 5200 | 7400 | 130.7(FWD) | 0.05LT. | 323 | 0.003048 |

NOTES: 1. OPEN AREA PRESENT WIND & AZIMUTH COMBINATIONS THAT YIELD LESS THAN 10% CONTROL MARGINS
2. DASHED LINES INDICATE EXTRPOLATED CURVES
3. SKID HEIGHT = 5 TO 15 FEET
4. DATA POINTS WERE DERIVED FROM FIGURES 32 THROUGH 38

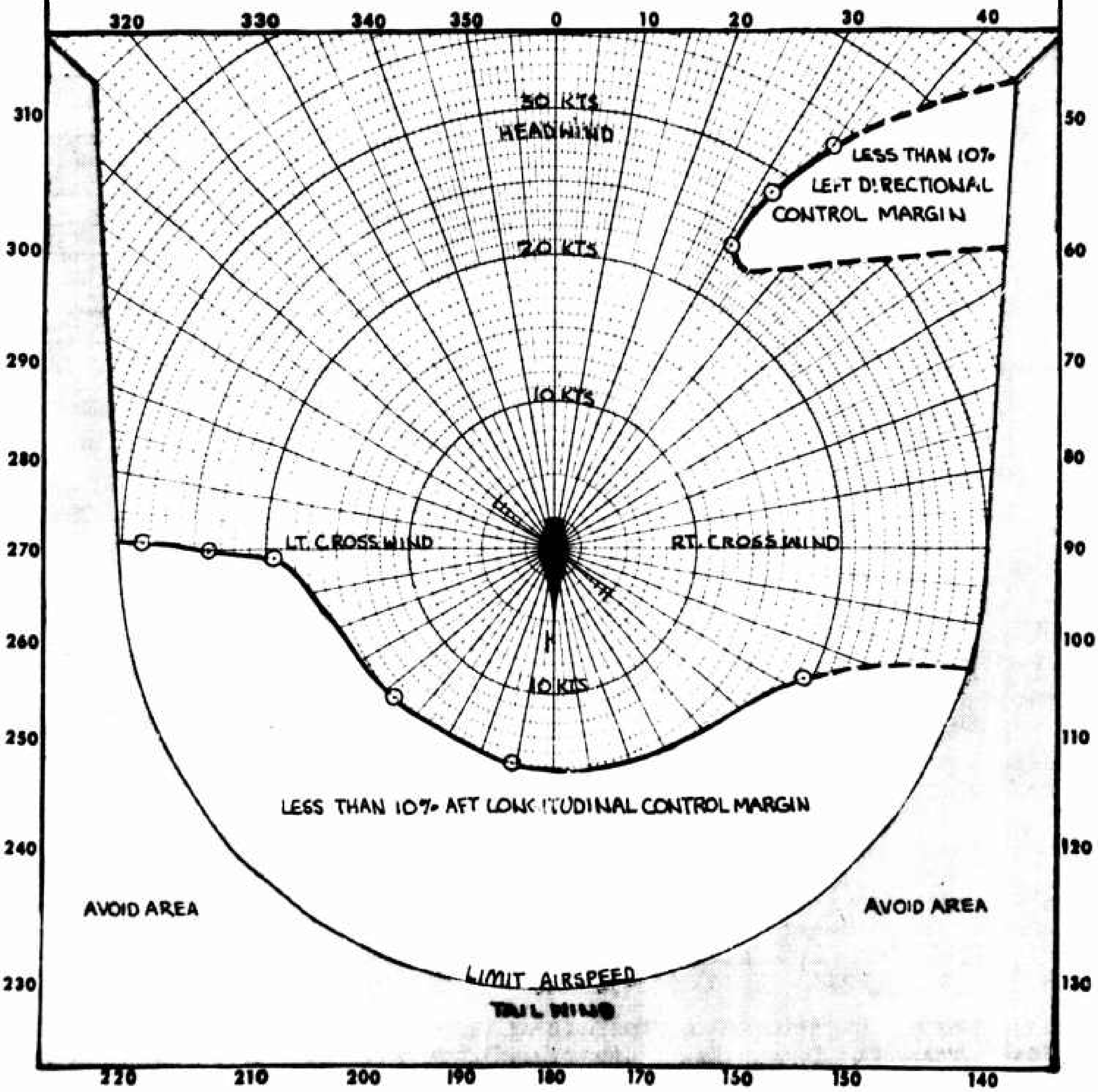


FIGURE 28

TRANSLATIONAL FLIGHT CONTROL MARGIN SUMMARY

UH-1H USA S/N 6717145

| AVG. DENSITY ALTITUDE ~ FEET | AVG. GRWT ~LB | AVG. LONG. C.G. ~IN | AVG. LAT. C.G. ~IN. | AVG. ROTOR SPEED ~RPM | AVG. THRUST COEFF. – $C_T$ |
|---|---|---|---|---|---|
| 5100 | 8600 | 130.2 (FWD) | 0.04 LT. | 322 | 0.003554 |

NOTES: 1. OPEN AREA PRESENT WIND & AZIMUTH COMBINATIONS THAT YIELD LESS THAN 10% CONTROL MARGINS
2. DASHED LINES INDICATE EXTRAPOLATED CURVES
3. SKID HEIGHT: 5 TO 15 FEET
4. DATA POINTS WERE DERIVED FROM FIGURES 39 THROUGH 44

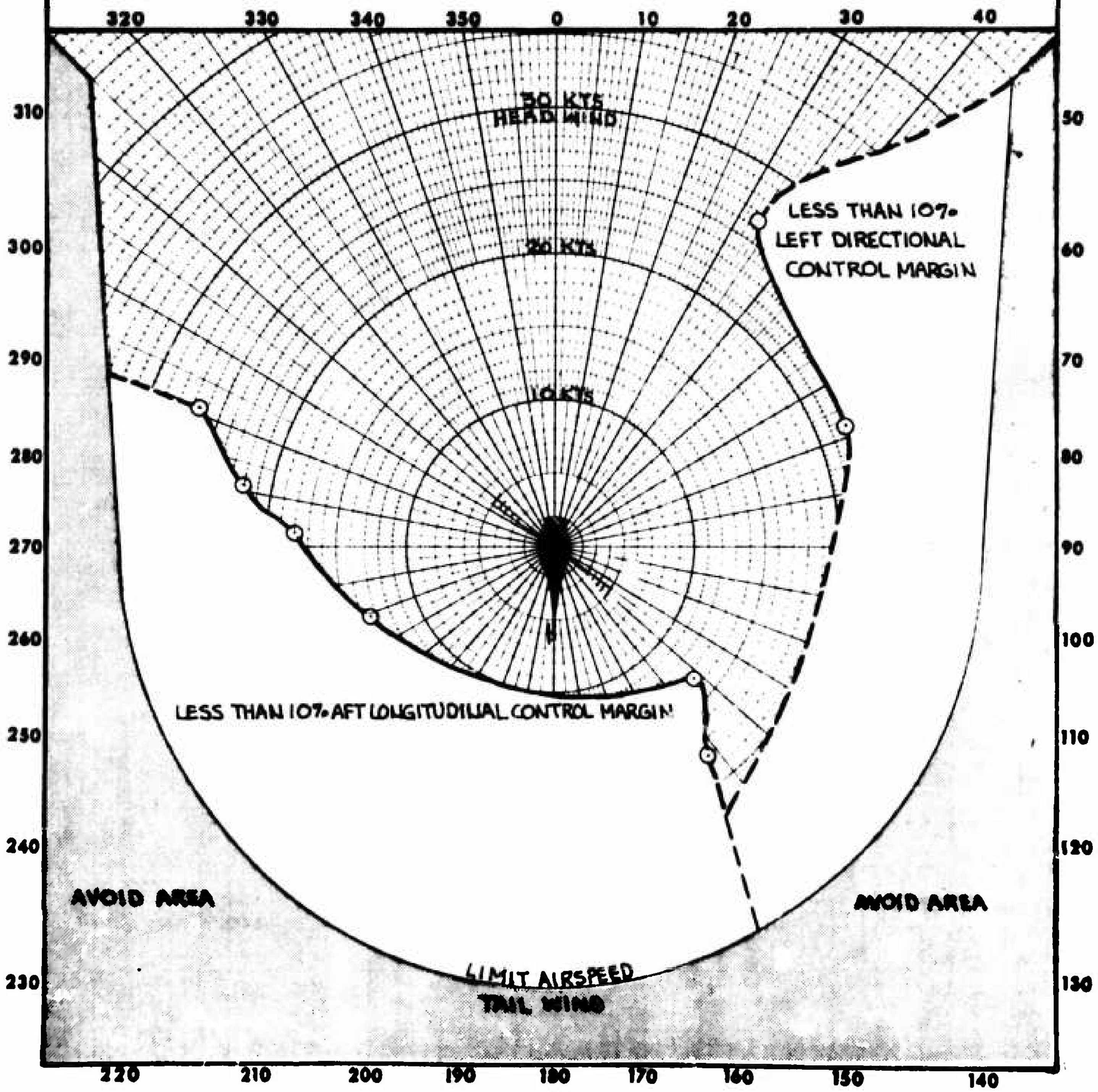


FIGURE 29

TRANSLATIONAL FLIGHT CONTROL MARGIN SUMMARY

UH-1H USA S/N 6717145

| AVG. DENSITY ALTITUDE ~FEET | AVG GRWT ~LB | AVG. LONG. C.G. ~IN. | AVG. LAT. C.G ~IN. | AVG. ROTOR SPEED ~RPM | AVG. THRUST COEFF. ~ $C_T$ |
|---|---|---|---|---|---|
| 11300 | 7600 | 130.5 (FWD) | 0.05 LT. | 324 | 0.003759 |

NOTES: 1. OPEN AREA PRESENT WIND & AZIMUTH COMBINATIONS THAT YIELD LESS THAN 10% CONTROL MARGINS

2. DASHED LINES INDICATE EXTRAPOLATED CURVES

3. SKID HEIGHT: 5 TO 15 FEET

4. DATA POINTS WERE DERIVED FROM FIGURES 45 THROUGH 51

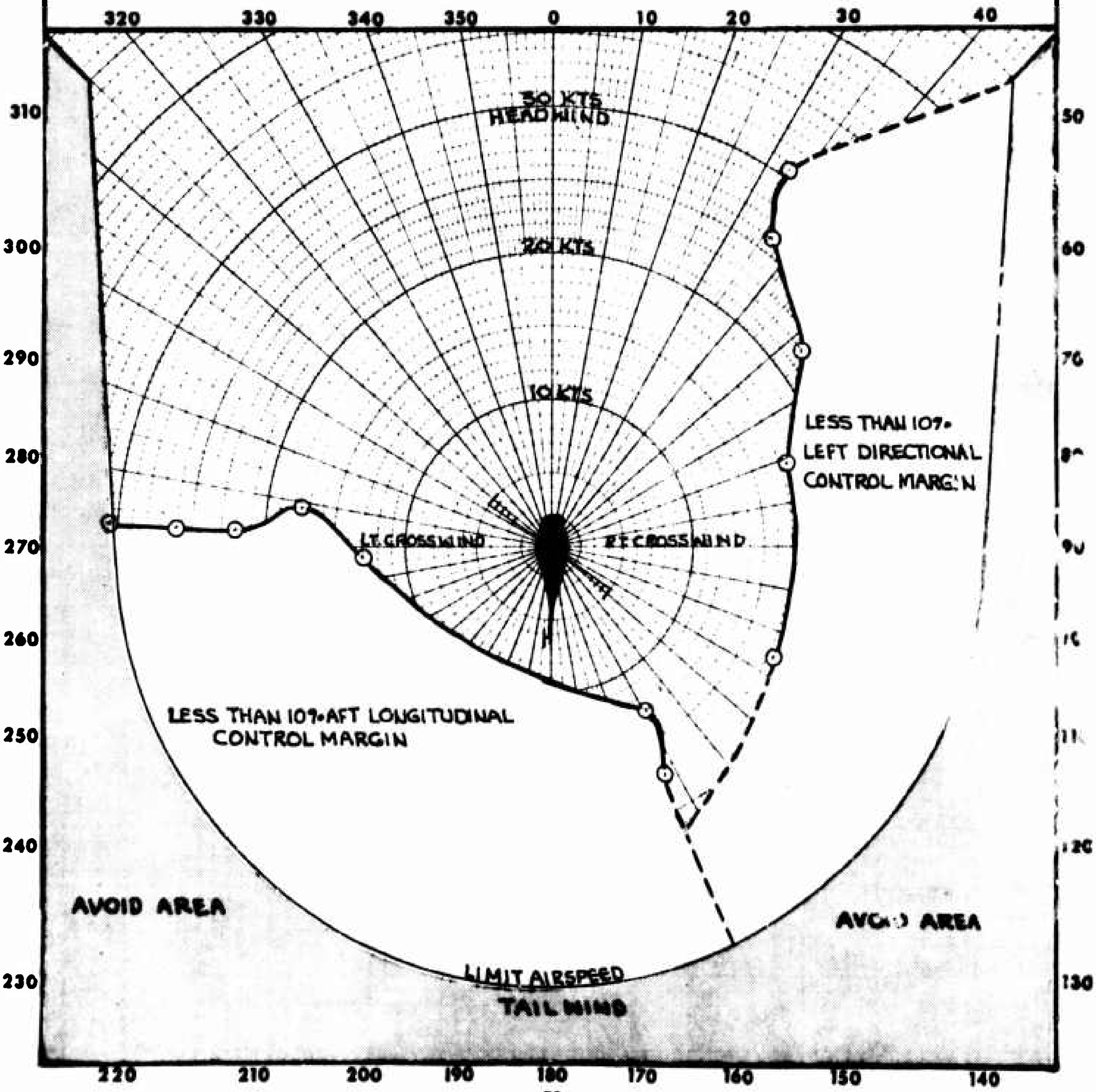


FIGURE 30

TRANSLATIONAL FLIGHT CONTROL MARGIN SUMMARY

UH-1H USA S/N 6717145

| AVG. DENSITY ALTITUDE ~ FEET | AVG. GRWT ~ LB. | AVG. LONG. C.G. ~ IN. | AVG. LAT. C.G. ~ IN. | AVG. ROTOR SPEED ~ RPM | AVG. THRUST COEFF. – $C_T$ |
|---|---|---|---|---|---|
| 5600 | 9400 | 133.9 (FWD) | 0.04 LT. | 323 | 0.003920 |

NOTES:
1. OPEN AREA PRESENT WIND & AZIMUTH COMBINATIONS THAT YIELD LESS THAN 10% CONTROL MARGINS
2. DASHED LINES INDICATE EXTRAPOLATED CURVES
3. SKID HEIGHT: 5 TO 15 FEET
4. DATA POINTS WERE DERIVED FROM FIGURES 52 THROUGH 57

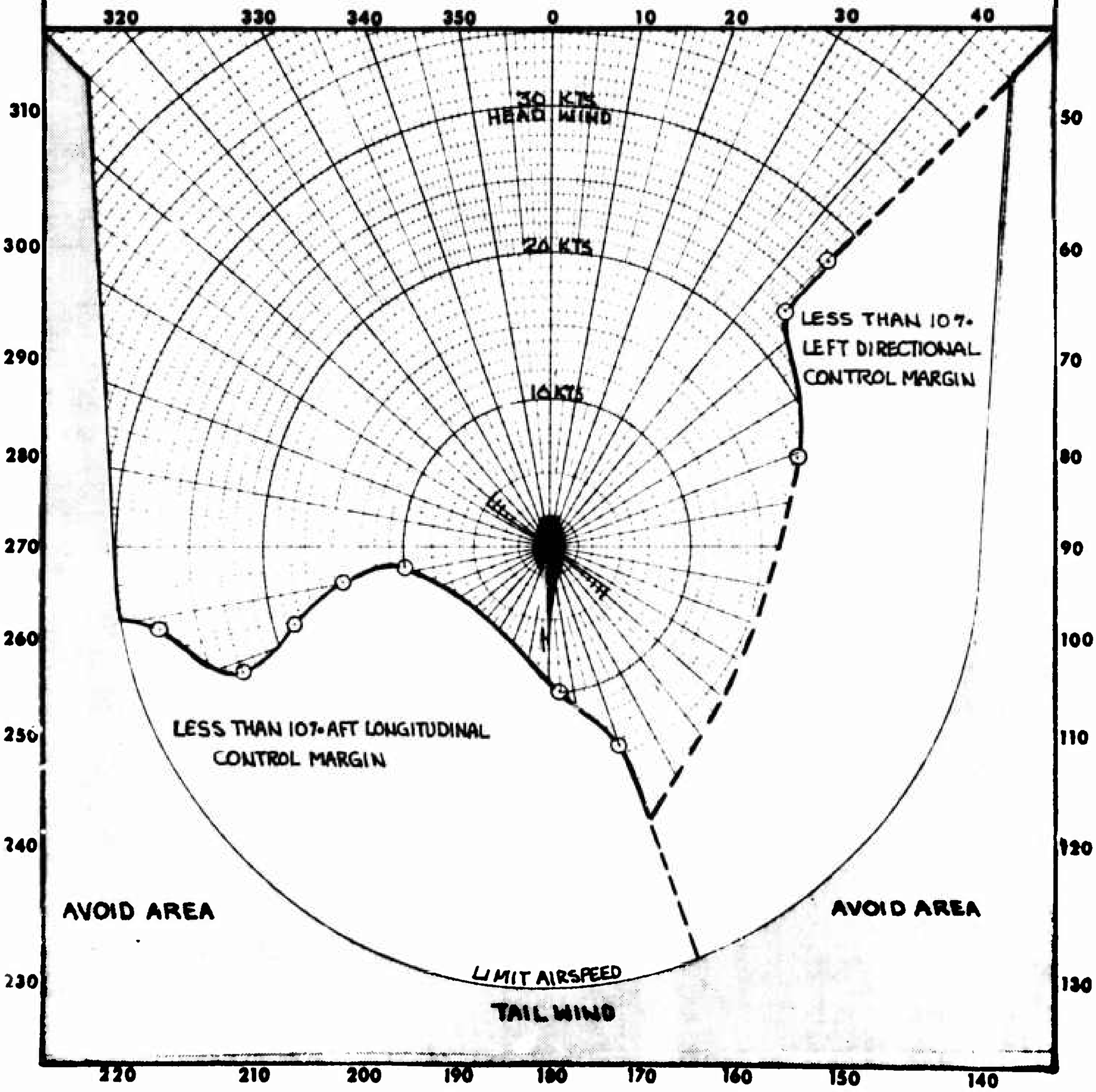


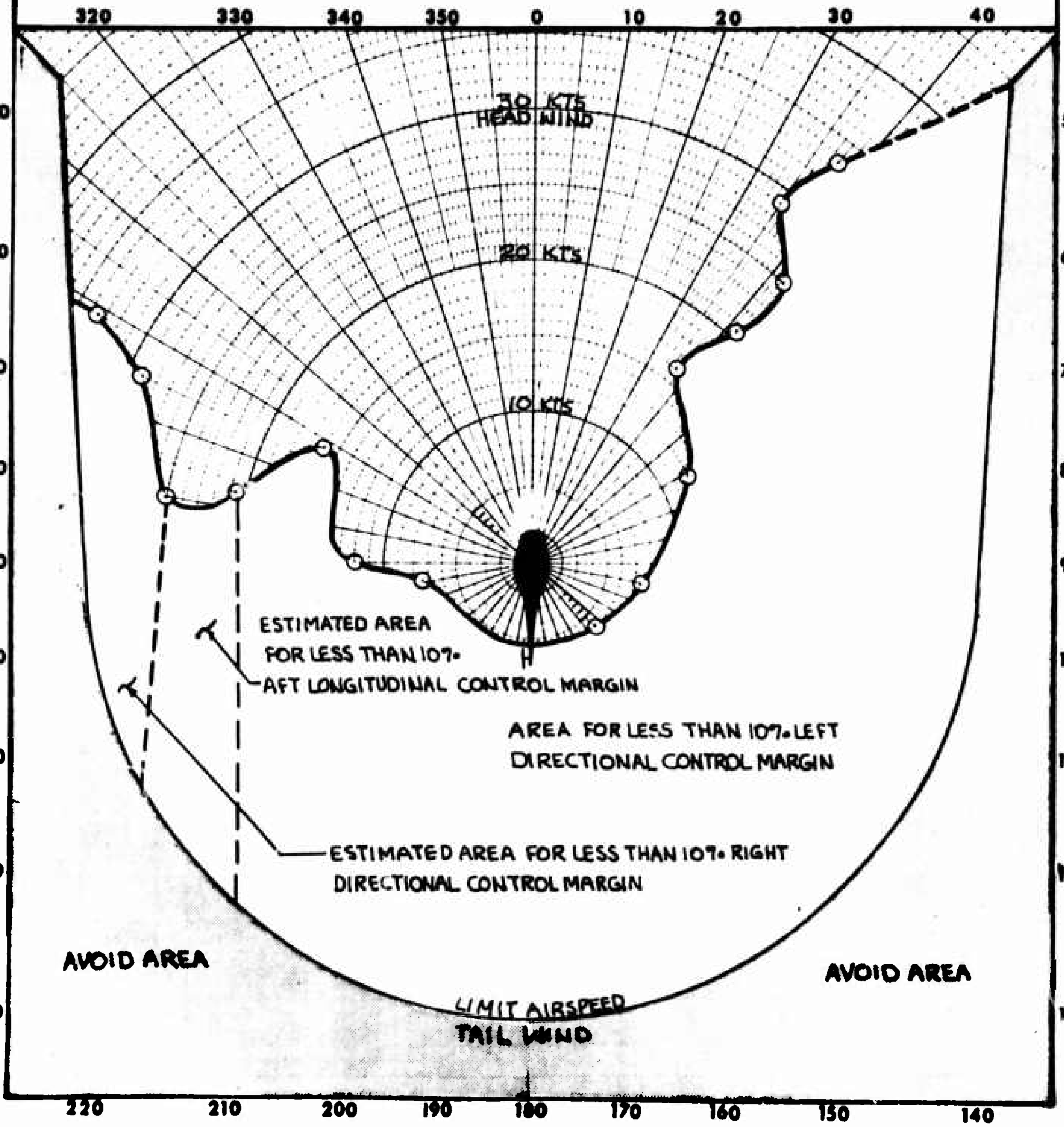
FIGURE 31

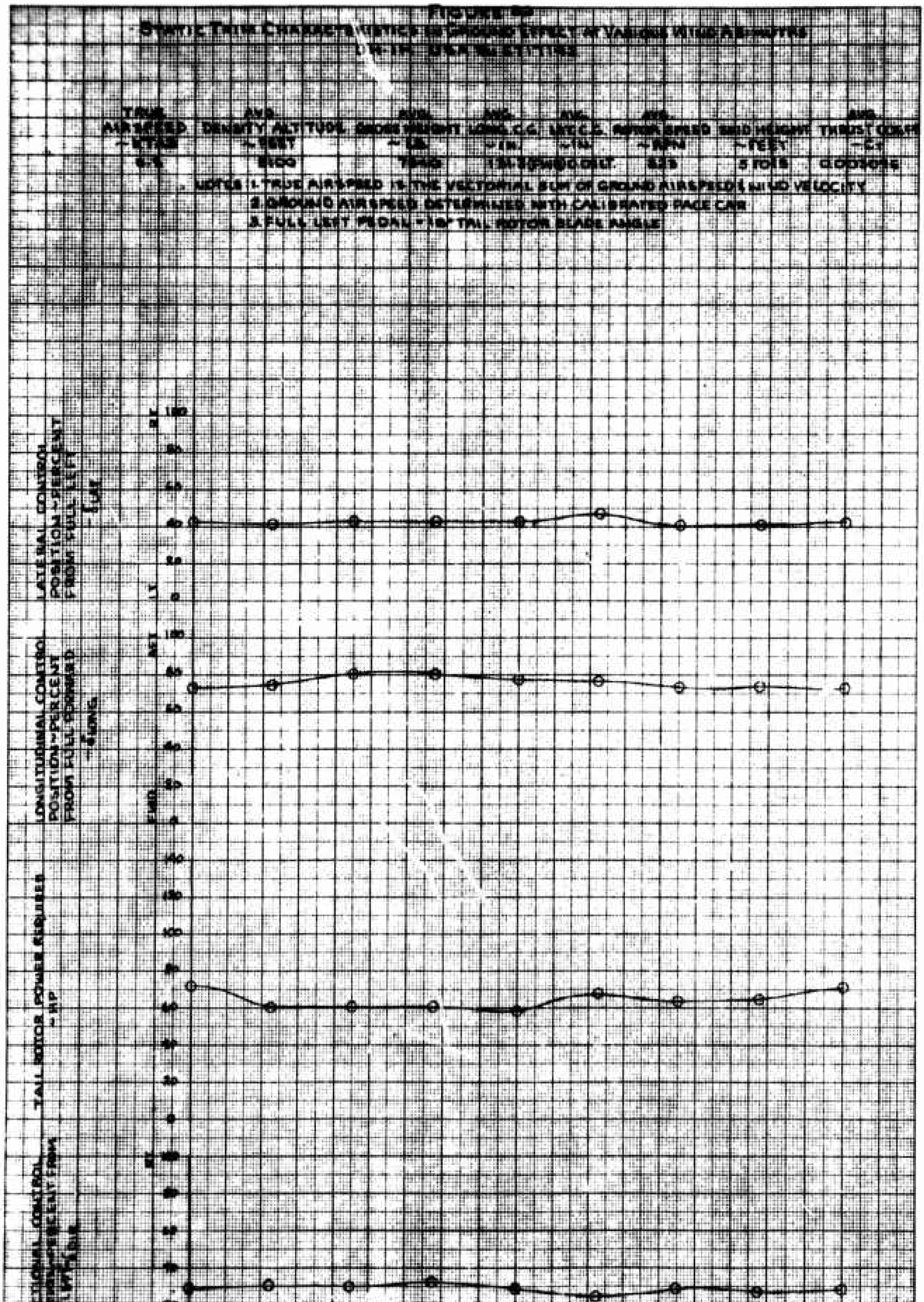
# TRANSLATIONAL FLIGHT CONTROL MARGIN SUMMARY

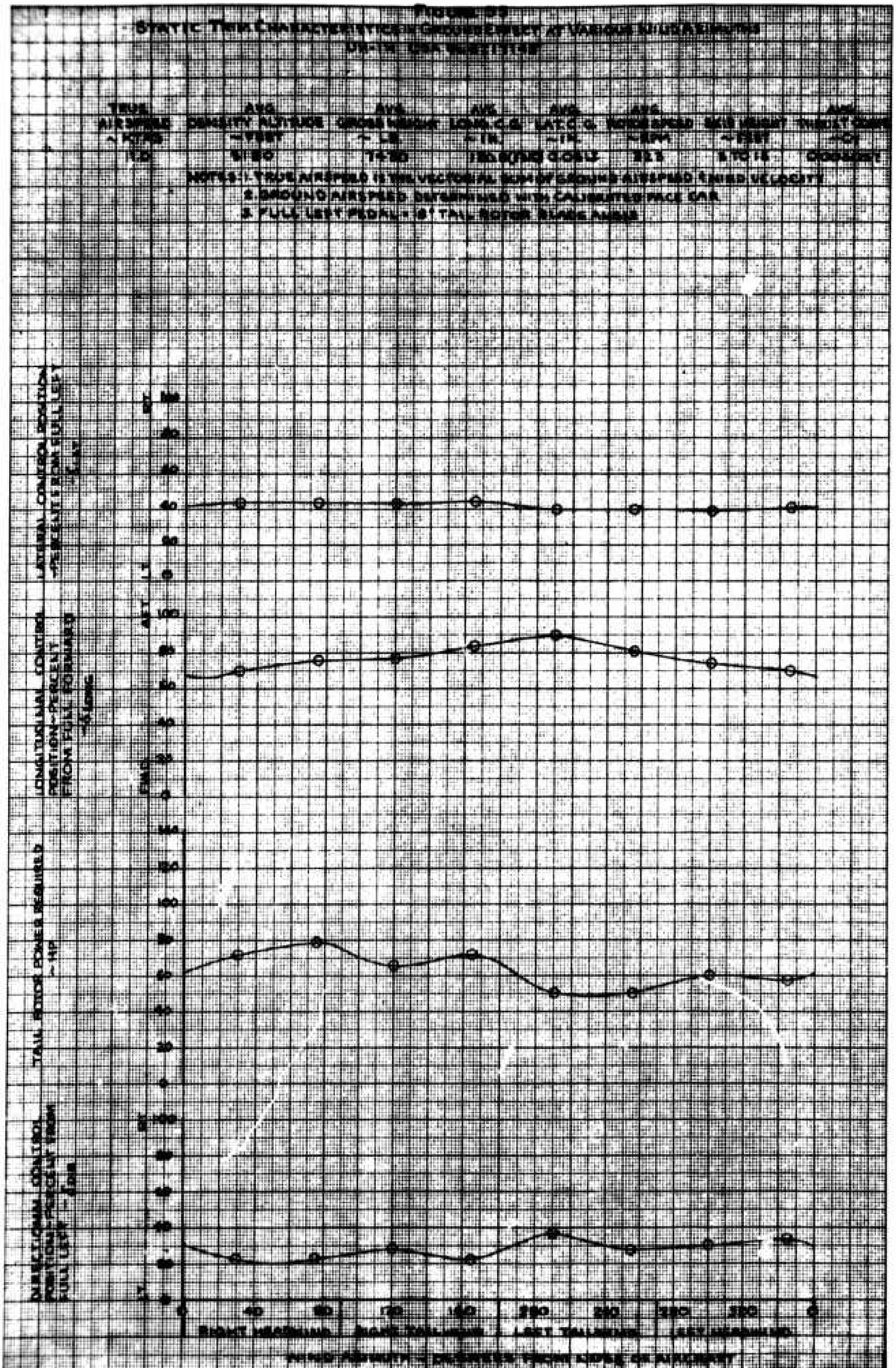
UH-1H USA S/N 6717145

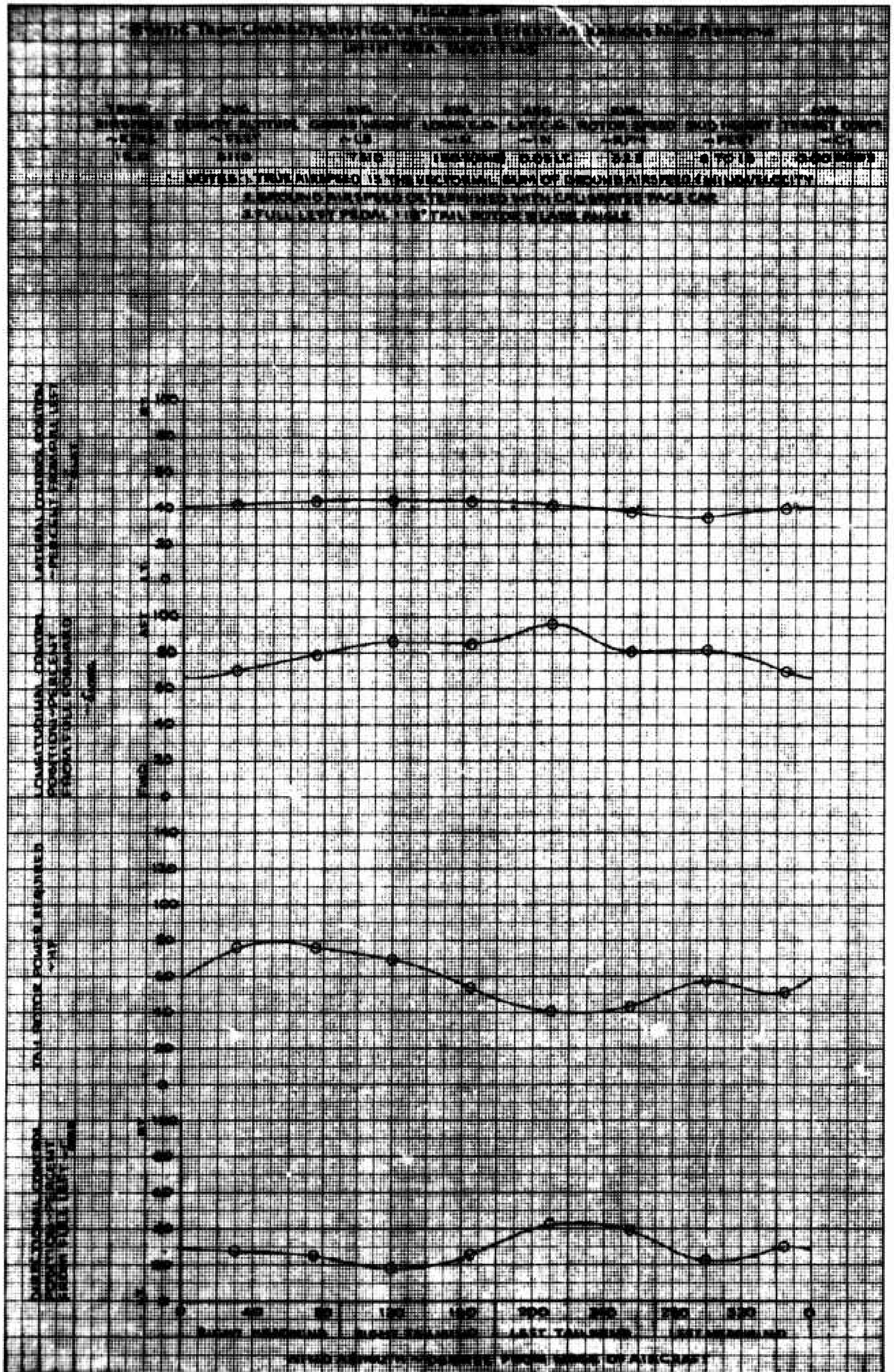
| AVG. DENSITY ALTITUDE ~FEET | AVG. GRWT ~LB. | AVG. LONG C.G. ~IN. | AVG. LAT. C.G. ~IN. | AVG. ROTOR SPEED ~RPM | AVG. THRUST COEFF. - $C_T$ |
|---|---|---|---|---|---|
| 11500 | 8600 | 130.5(FWD) | 0.04 LT. | 324 | 0.004281 |

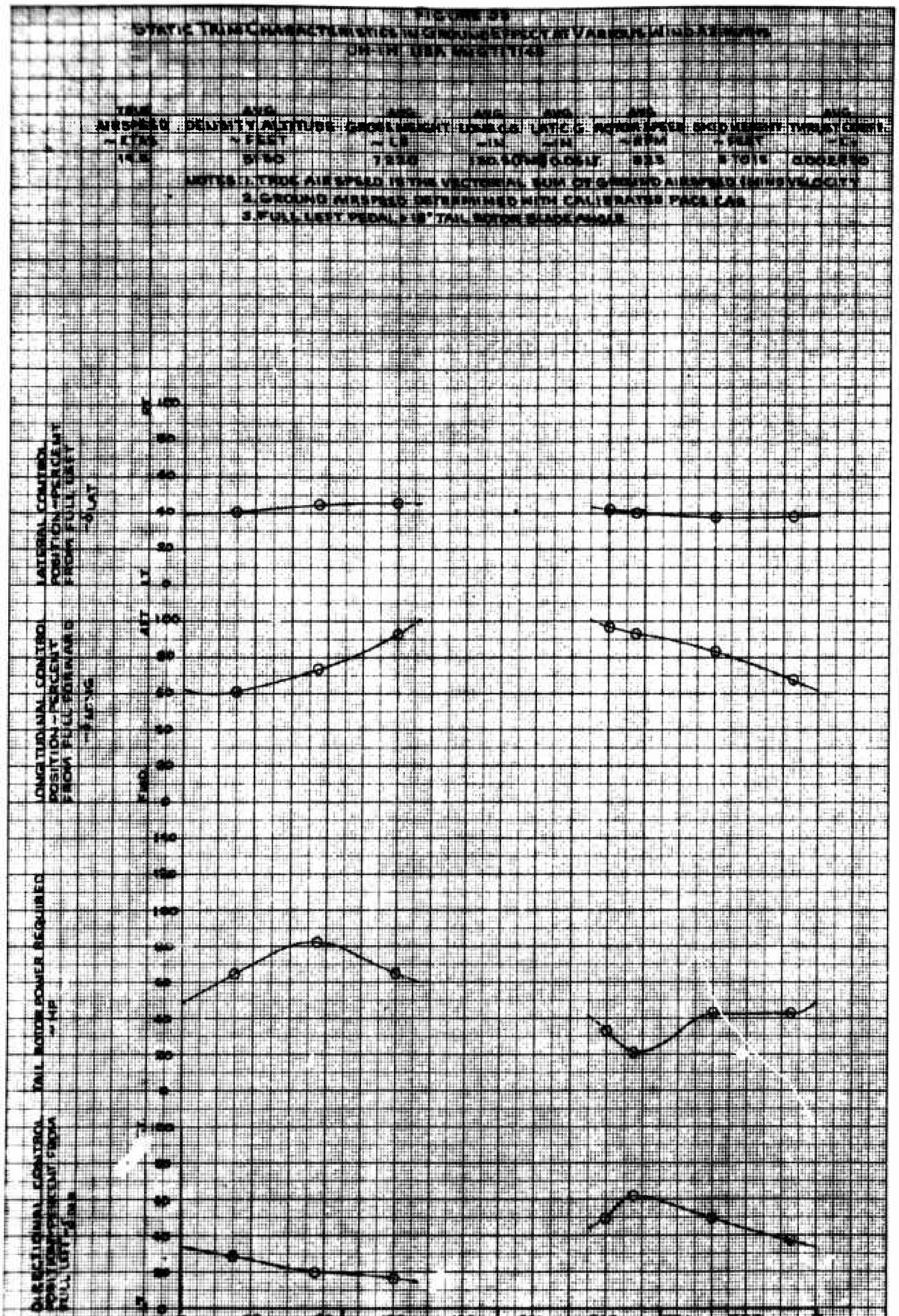
NOTES: 1. OPEN AREA PRESENT WIND & AZIMUTH COMBINATIONS THAT YIELD LESS THAN 10% CONTROL MARGIN
2. DASHED LINES INDICATE EXTRAPOLATED CURVES
3. SKID HEIGHT: 5 TO 15 FEET
4. DATA POINTS DERIVED FROM FIGURES 58 THROUGH 66



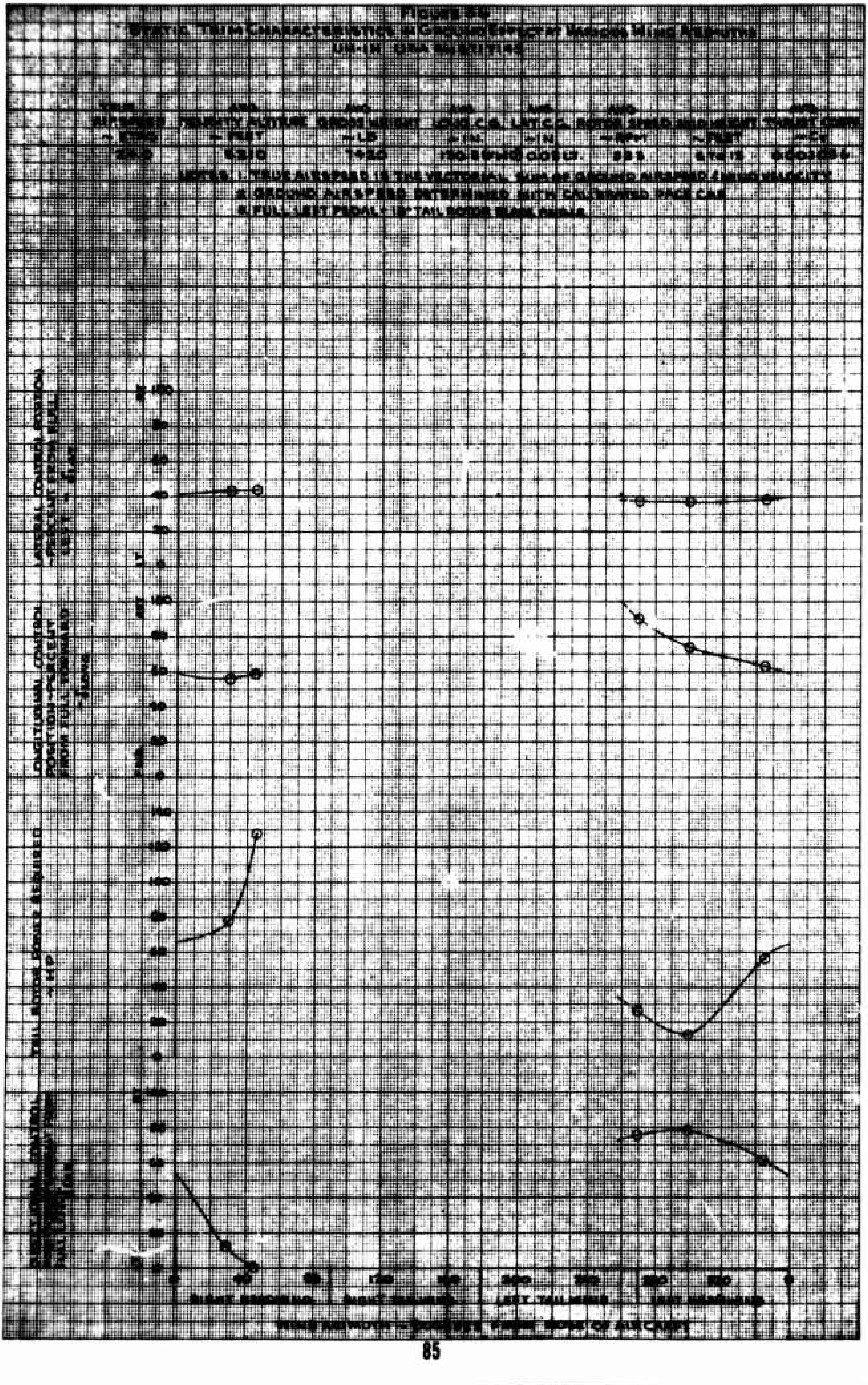

AVG. DENSITY ALTITUDE ~ FEET
AVG. GROSS WEIGHT ~ LB.
AVG. ROTOR SPEED ~ RPM
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED & WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL
LATERAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
0 40 80 120 160 200 240 280 320 0
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

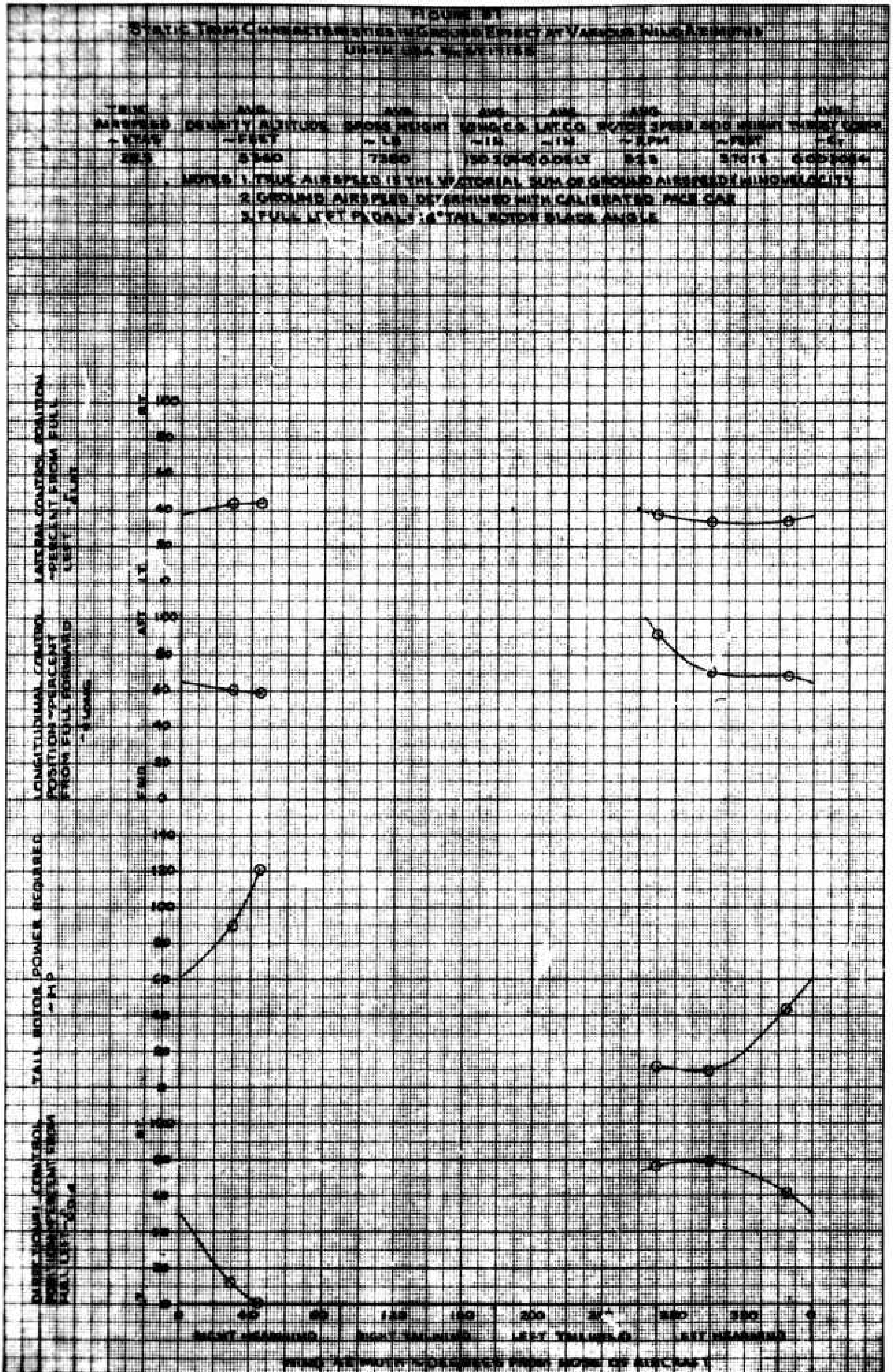

TRUE AIRSPEED ~KTAS
AVG. DENSITY ALTITUDE ~FEET
AVG. GROSS WEIGHT ~LB.
AVG. LONG. C.G. ~IN.
AVG. LAT. C.G. ~IN.
SKID HEIGHT ~FEET
LATERAL CONTROL POSITION
LONGITUDINAL CONTROL POSITION-PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~HP
DIRECTIONAL CONTROL POSITION-PERCENT FROM FULL LEFT
0
40
80
120
160
200
240
280
320
0


0 40 80 120 160 200 240 280 320 0

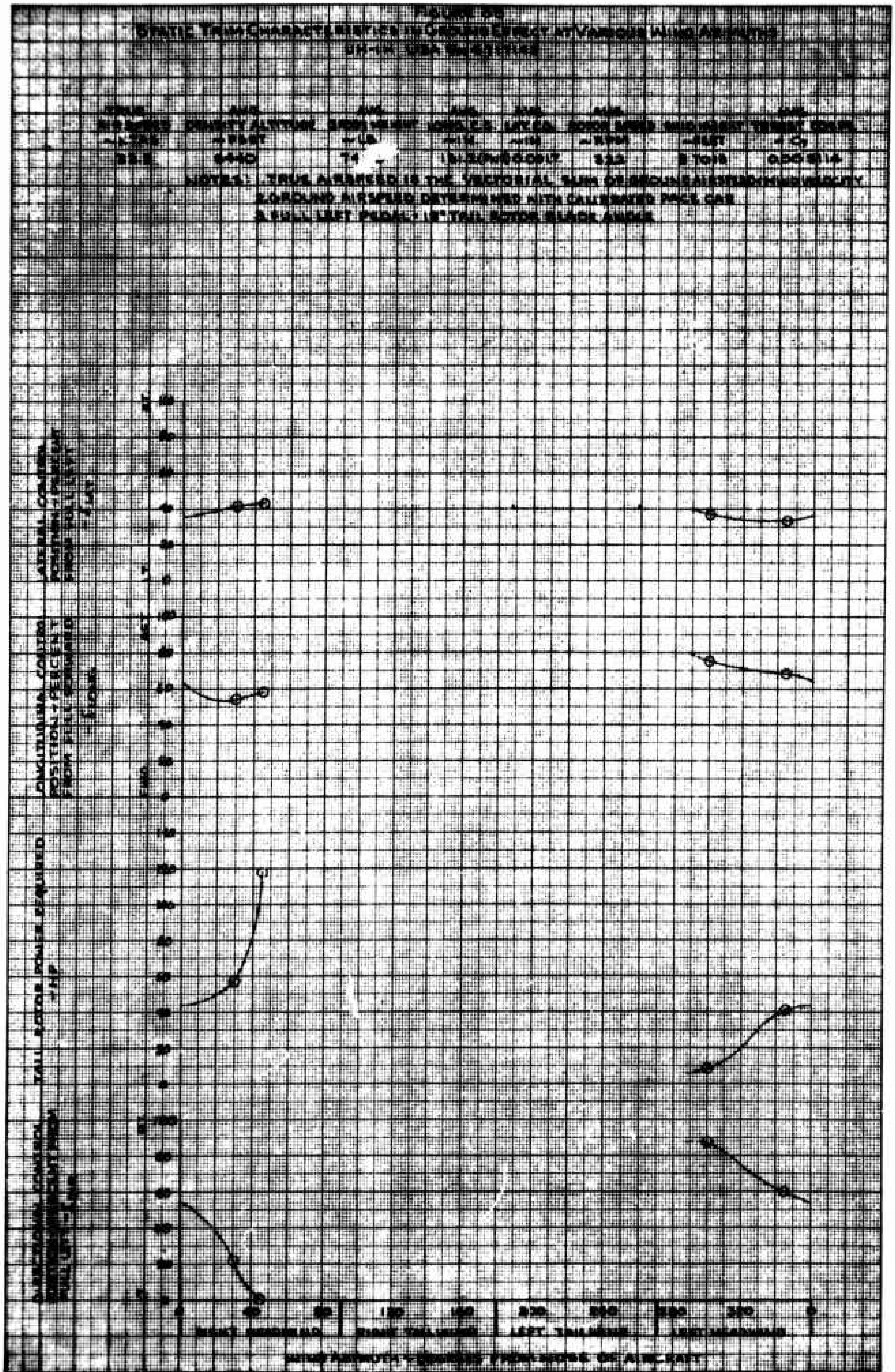


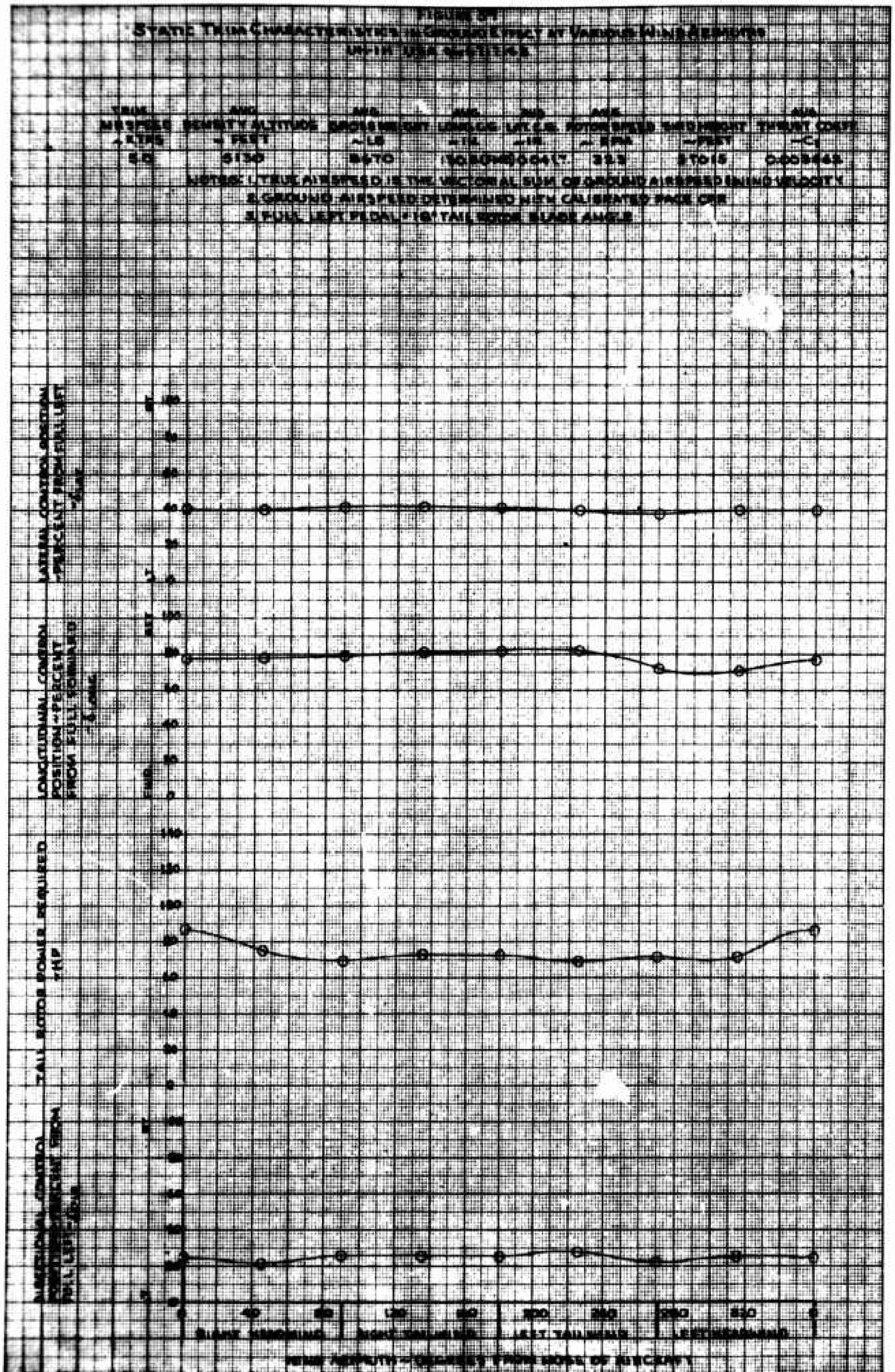
UH-1H USA
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
LATERAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
0
40
80
120
160
200
240
280
320
0
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

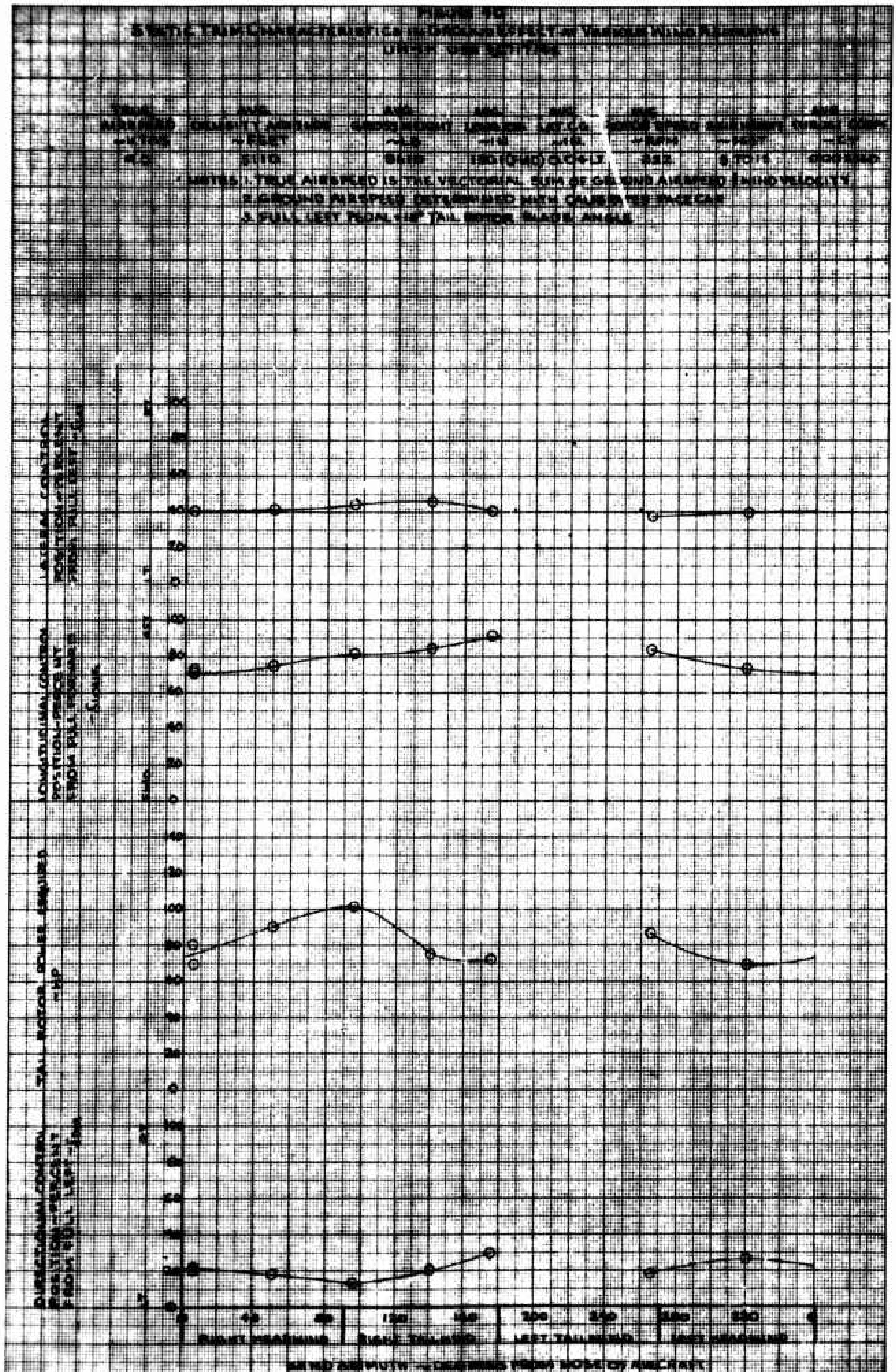

STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = 18° TAIL ROTOR BLADE ANGLE

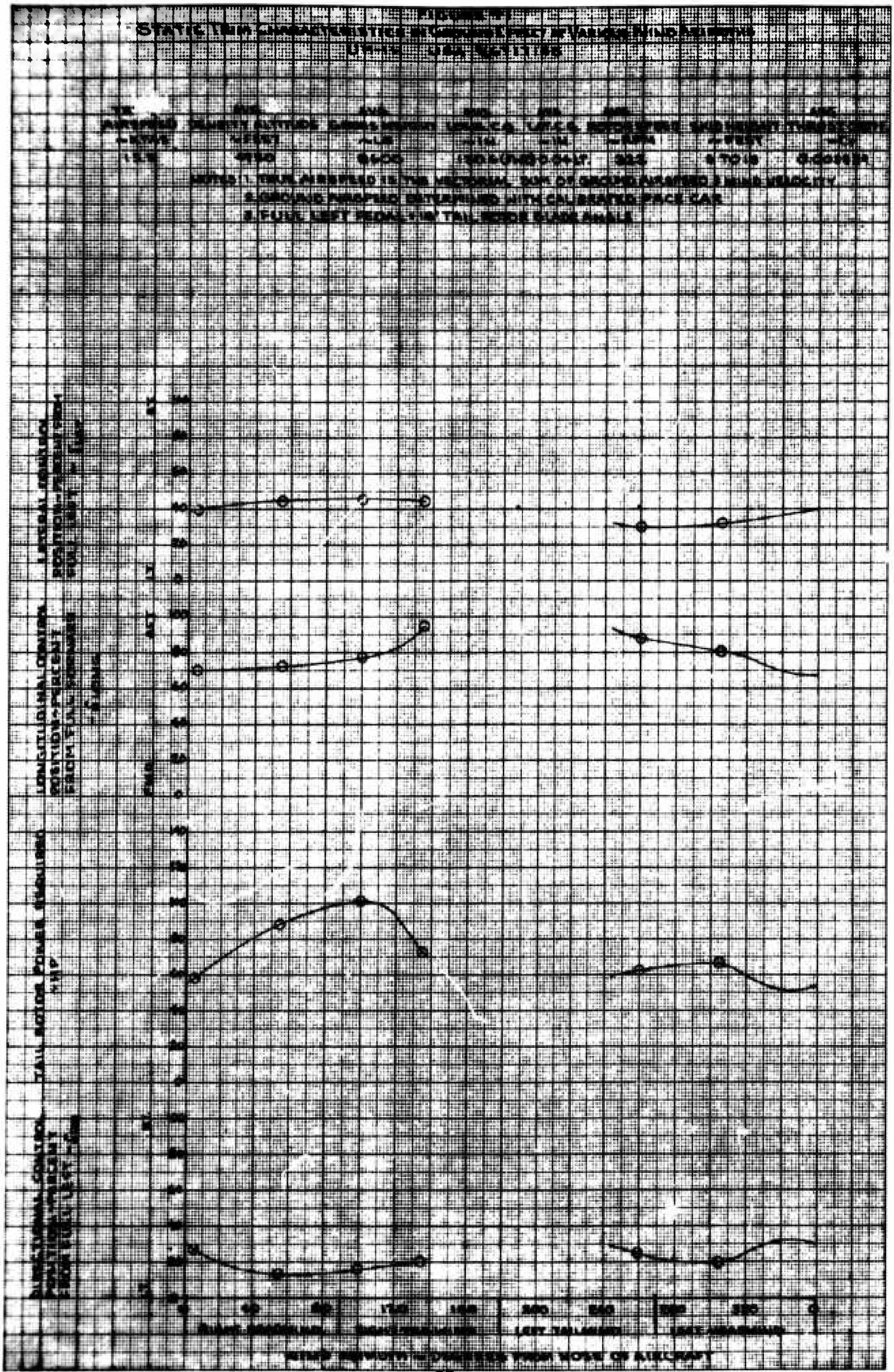


NOTES
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
TAIL ROTOR BLADE ANGLE
TAIL ROTOR POWER REQUIRED
~HP

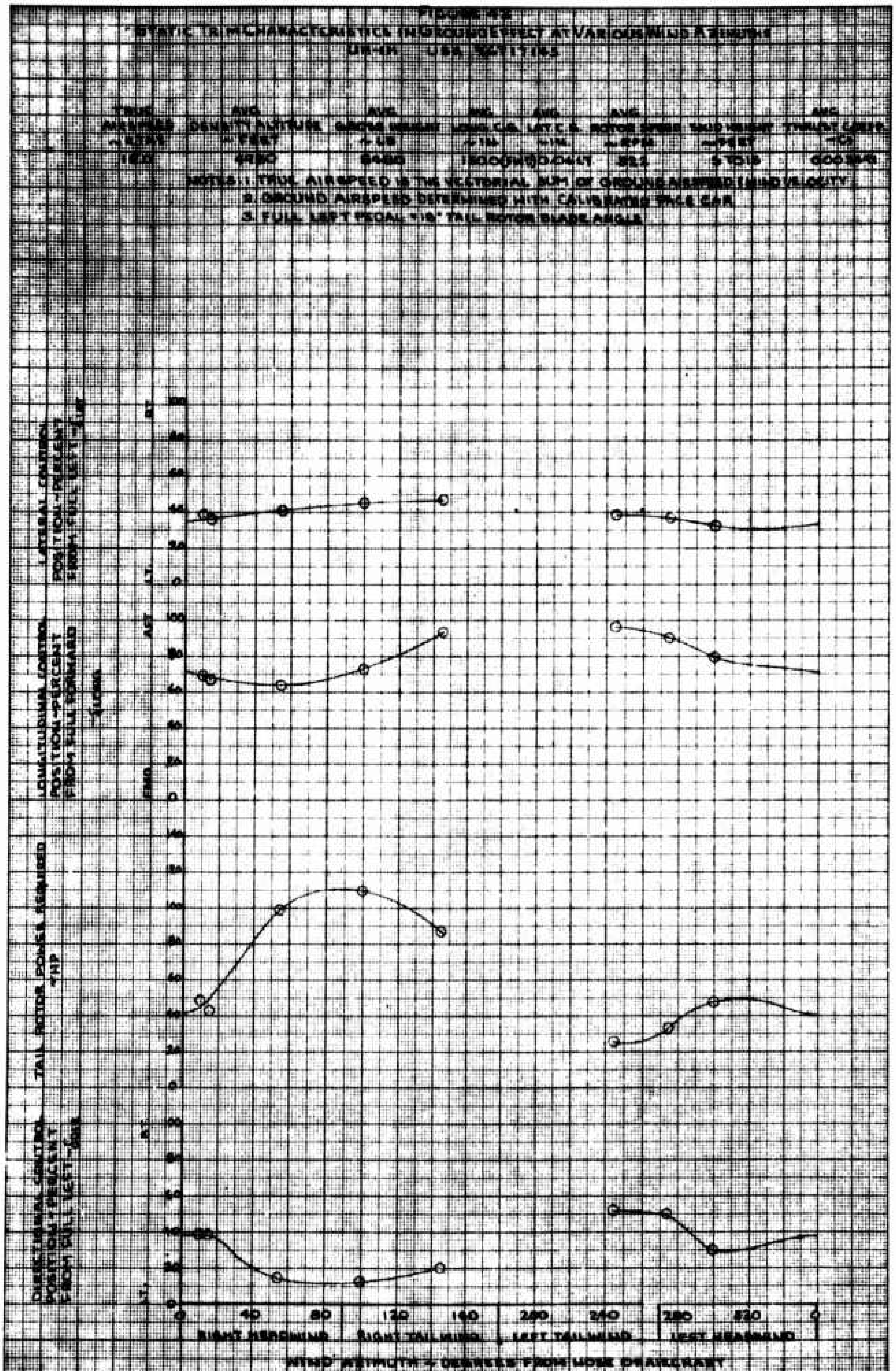


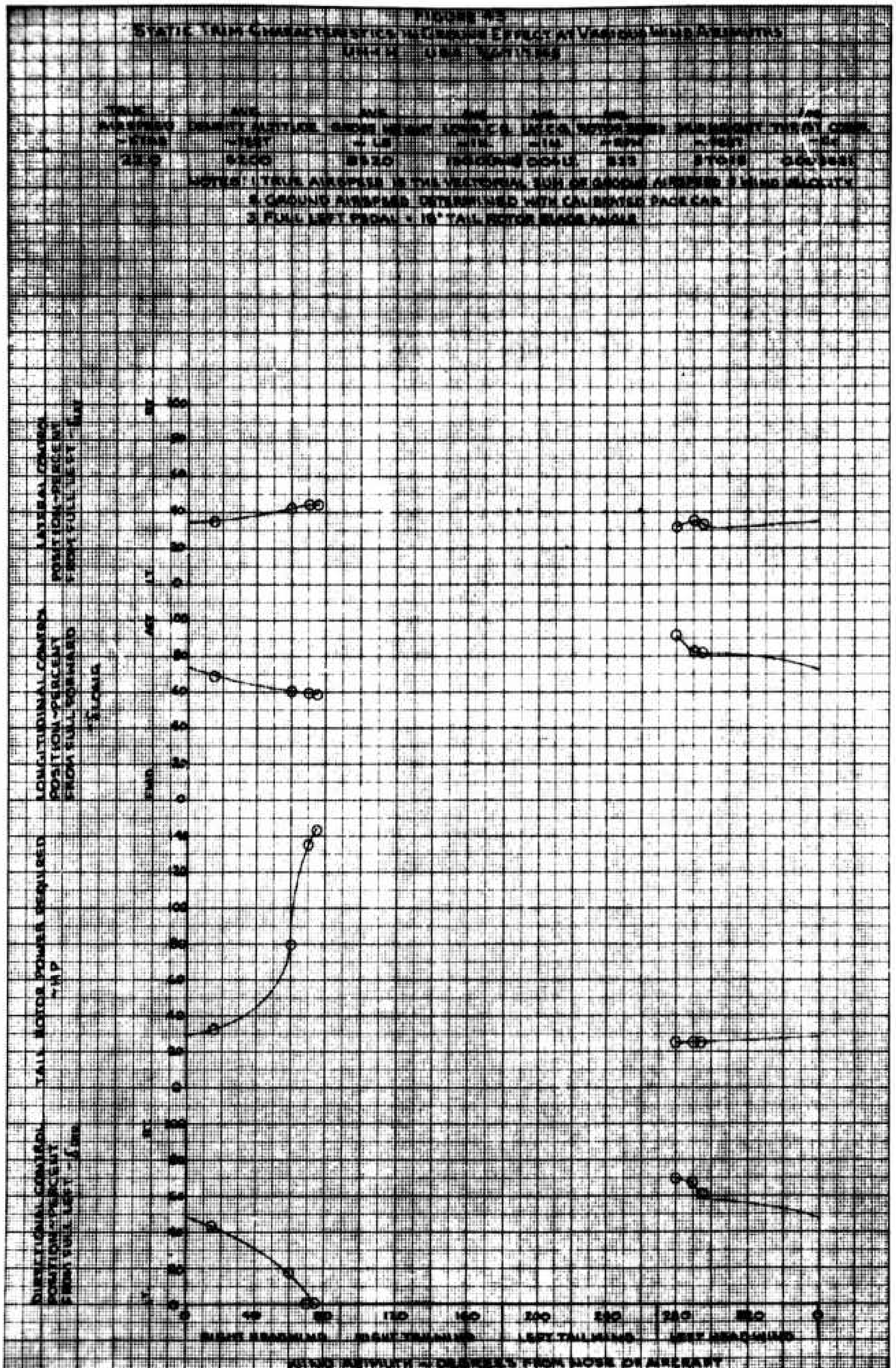

STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = 19° TAIL ROTOR BLADE ANGLE
LATERAL CONTROL POSITION ~PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~HP
DIRECTIONAL CONTROL POSITION ~PERCENT FROM FULL LEFT
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

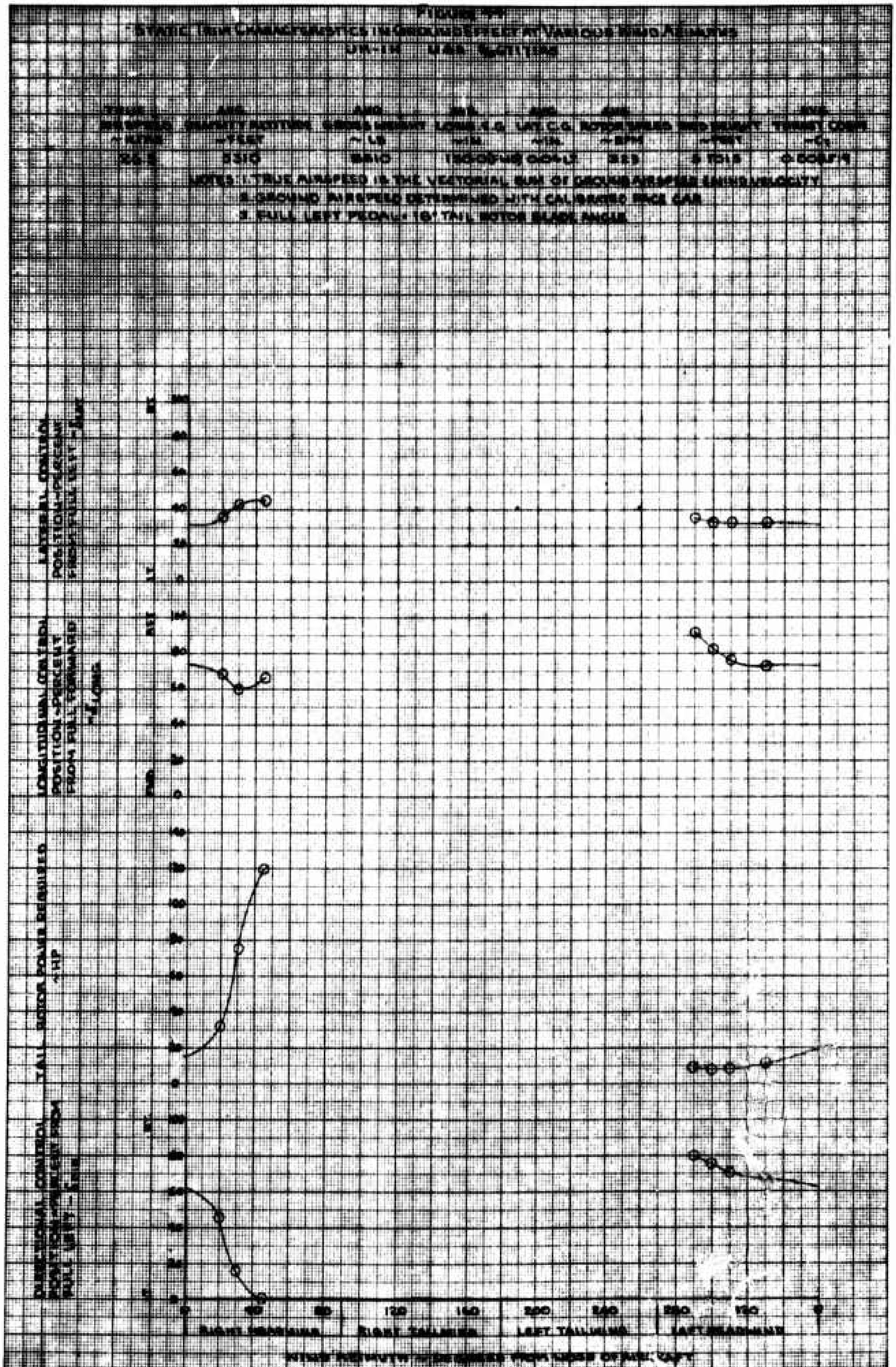

NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED & WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACECAR
DIRECTIONAL CONTROL POSITION~PERCENT FROM FULL LEFT
TAIL ROTOR POWER REQUIRED ~HP
LONGITUDINAL CONTROL POSITION~PERCENT FROM FULL FORWARD
LATERAL CONTROL POSITION~PERCENT FROM FULL LEFT
RIGHT CROSSWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT CROSSWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

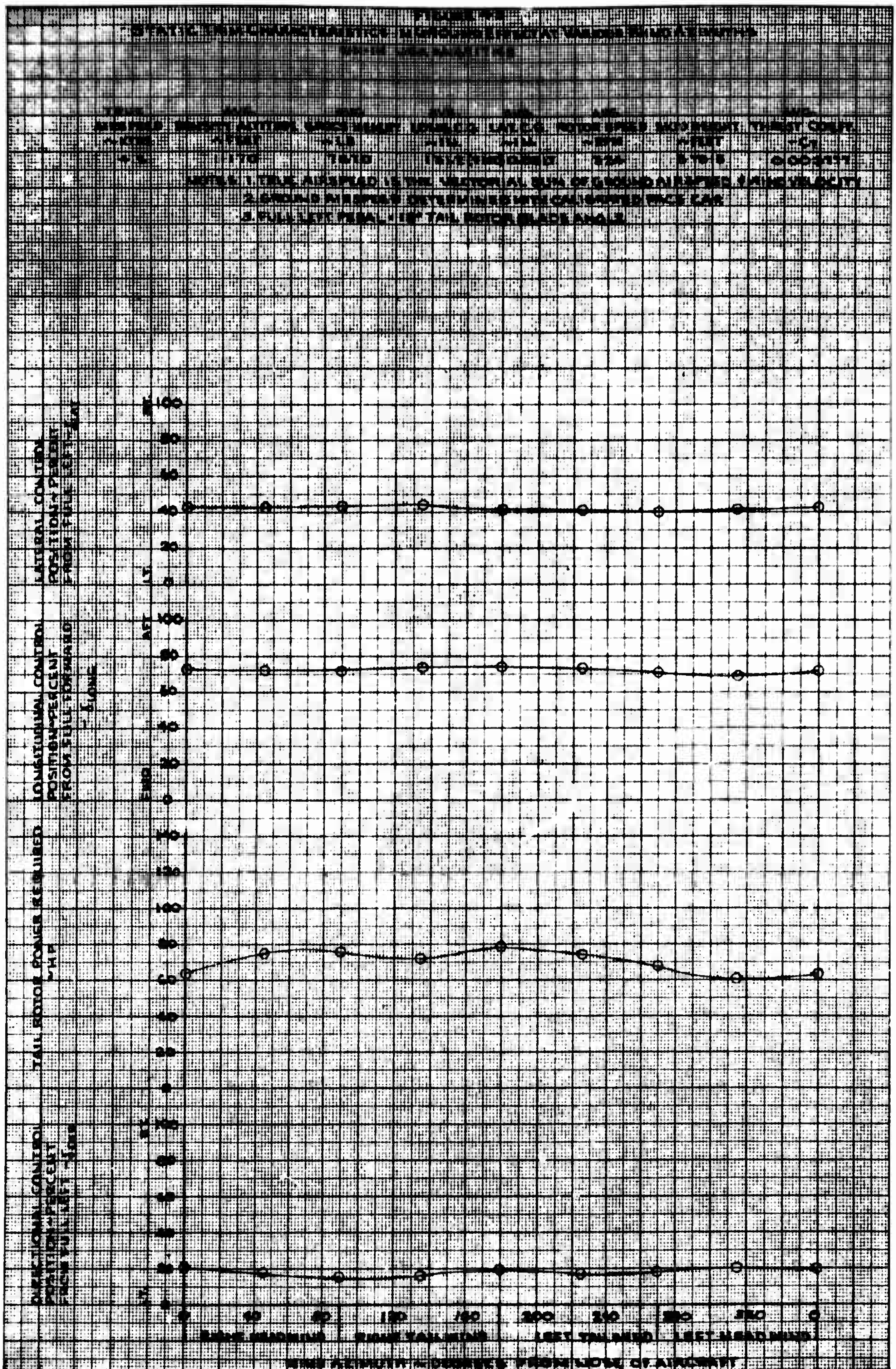


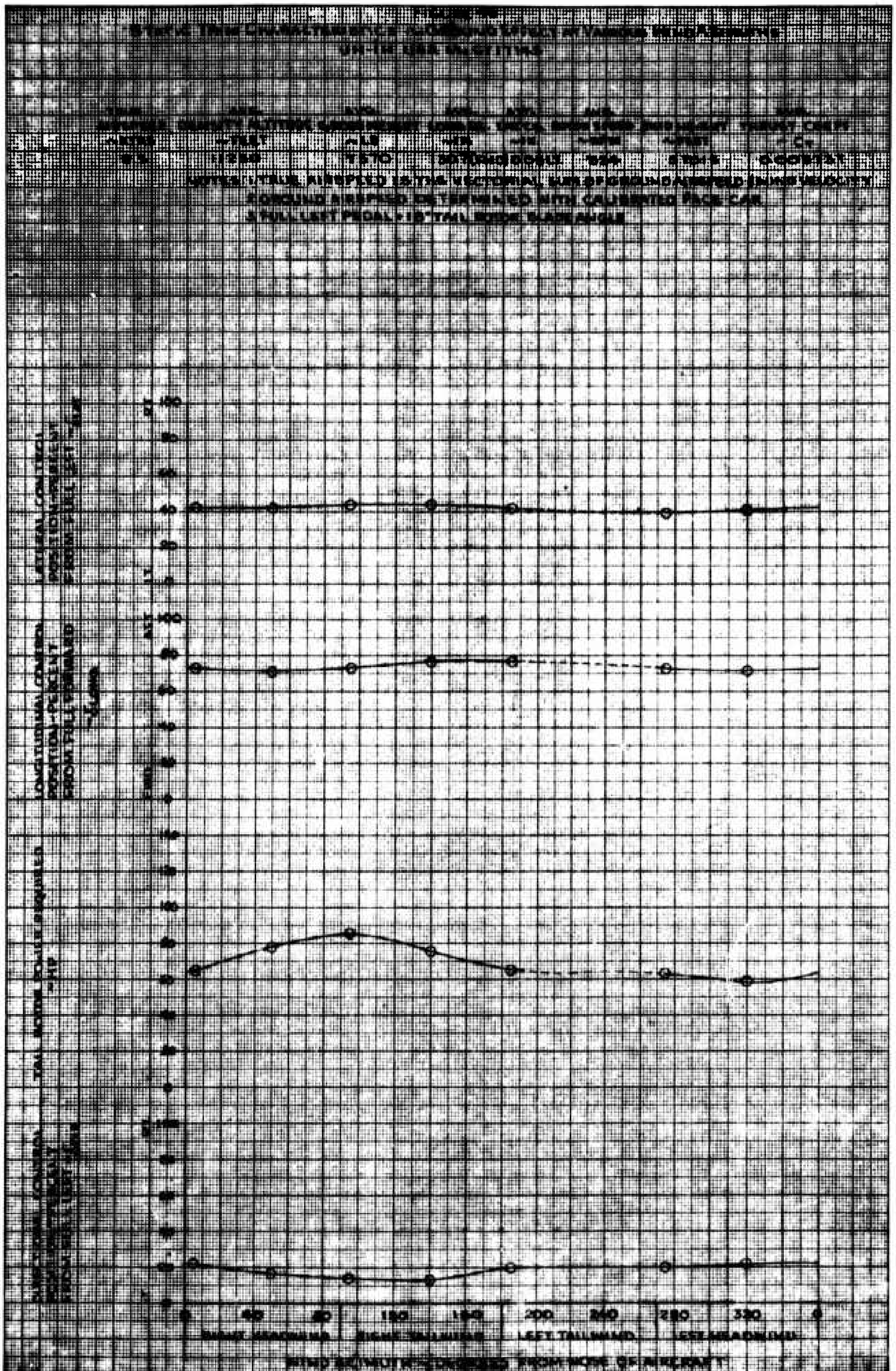
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL
TAIL ROTOR BLADE ANGLE

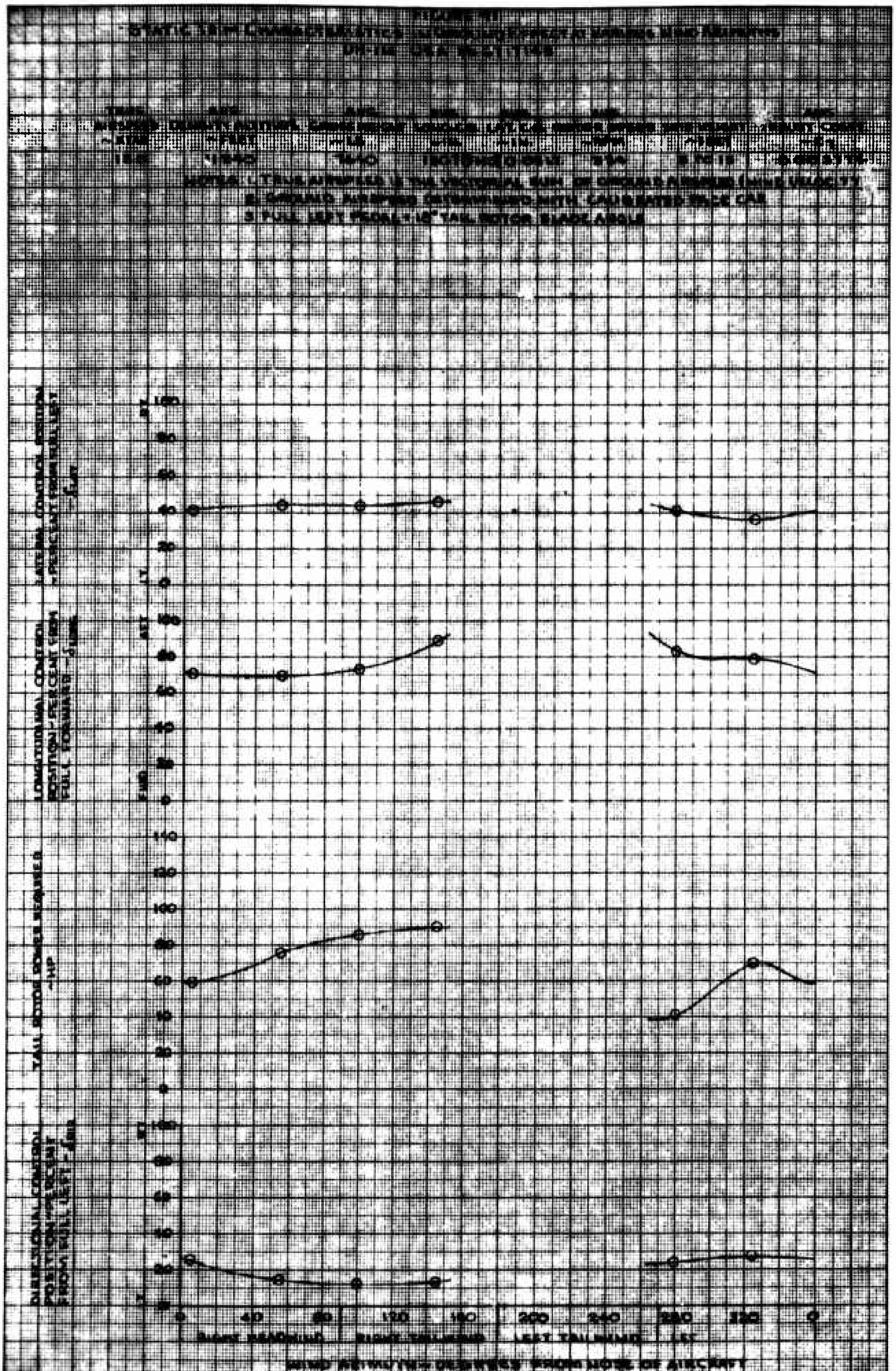

STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS
UH-1H USA S/N 17165
TRUE AIRSPEED ~KTAS
AVG. DENSITY ALTITUDE ~FEET
AVG. GROSS WEIGHT ~LB
AVG. LONG. C.G. ~IN.
AVG. LAT. C.G. ~IN.
AVG. ROTOR SPEED ~RPM
SKID HEIGHT ~FEET
AVG. THRUST COEFF. ~$C_T$
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL +19° TAIL ROTOR BLADE ANGLE
LATERAL CONTROL POSITION ~PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~HP
DIRECTIONAL CONTROL POSITION ~PERCENT FROM FULL LEFT
0
40
80
120
160
200
240
280
320
0
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

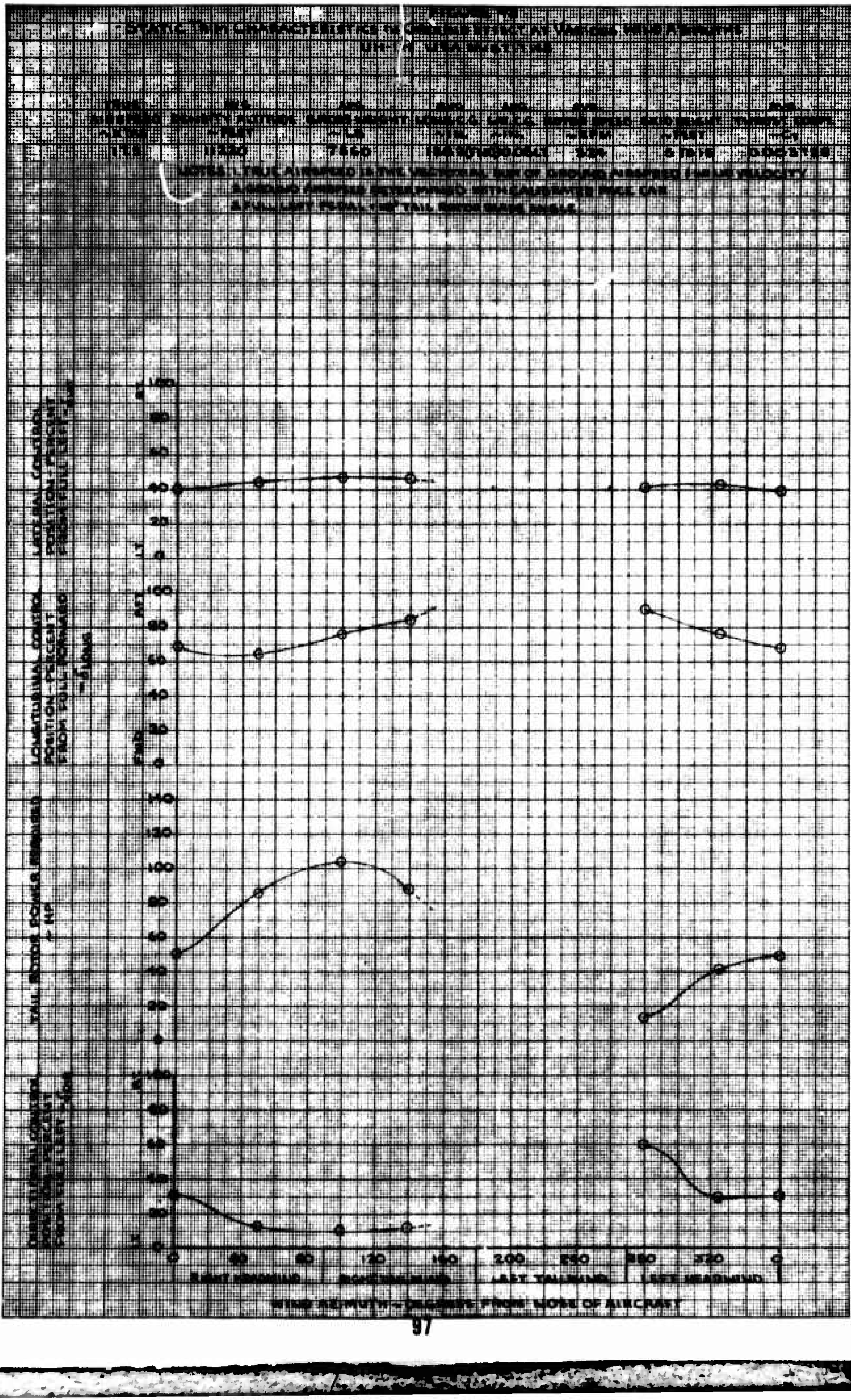

2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
TAIL ROTOR POWER REQUIRED ~ HP
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
LATERAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT


STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS
UH-1H
5310
8810
323
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = 16° TAIL ROTOR BLADE ANGLE
LATERAL CONTROL POSITION~PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION~PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~HP
DIRECTIONAL CONTROL POSITION~PERCENT FROM FULL LEFT
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT


LATERAL CONTROL
LONGITUDINAL CONTROL
TAIL ROTOR POWER REQUIRED
DIRECTIONAL CONTROL




STATIC TRIM CHARACTERISTICS
TAIL ROTOR POWER REQUIRED ~HP
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT
0
40
80
120
160
200
240
280
320
0


LATERAL CONTROL POSITION - PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION - PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION - PERCENT FROM FULL LEFT
0
40
80
120
160
200
240
280
320
0
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

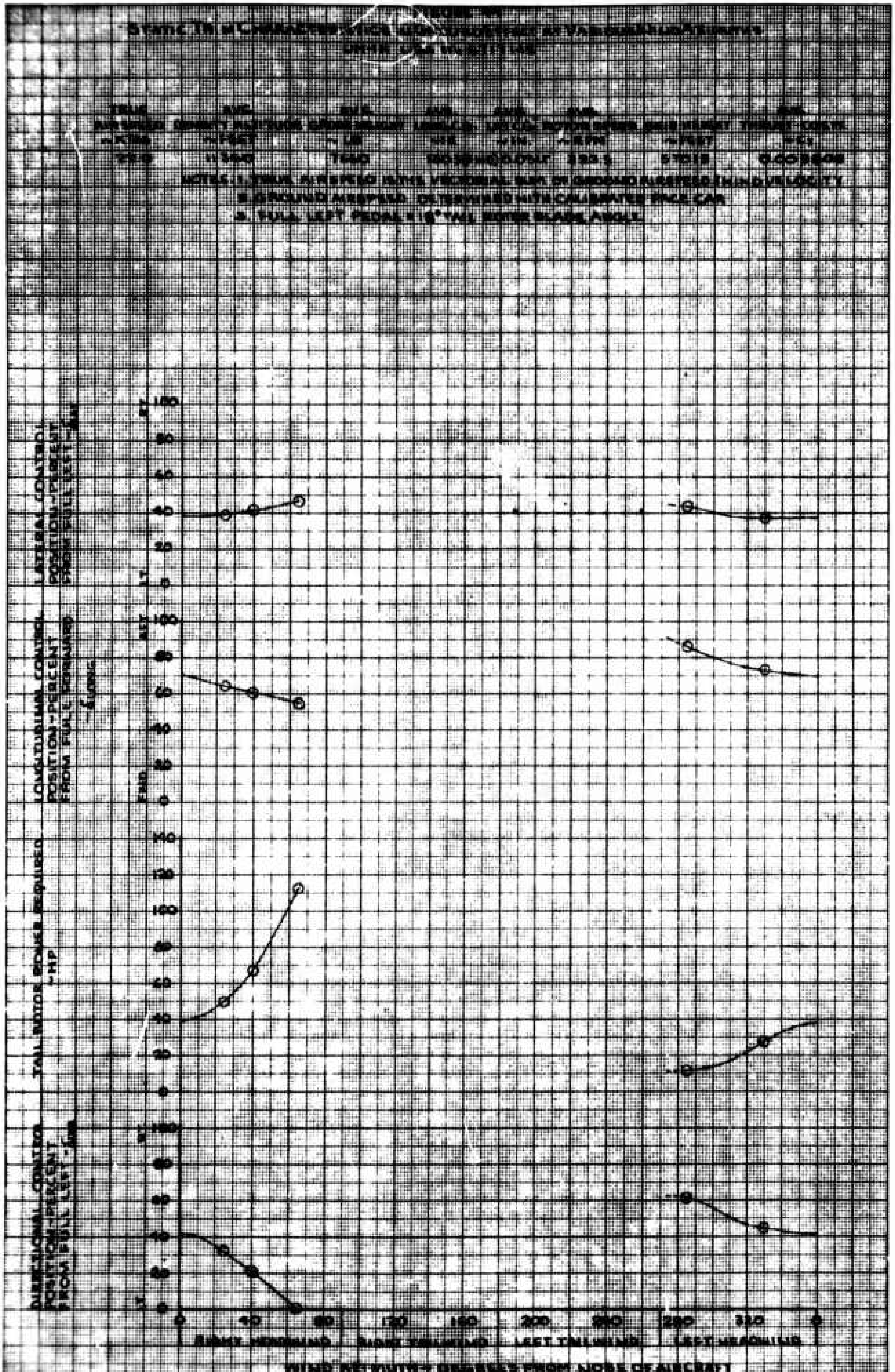
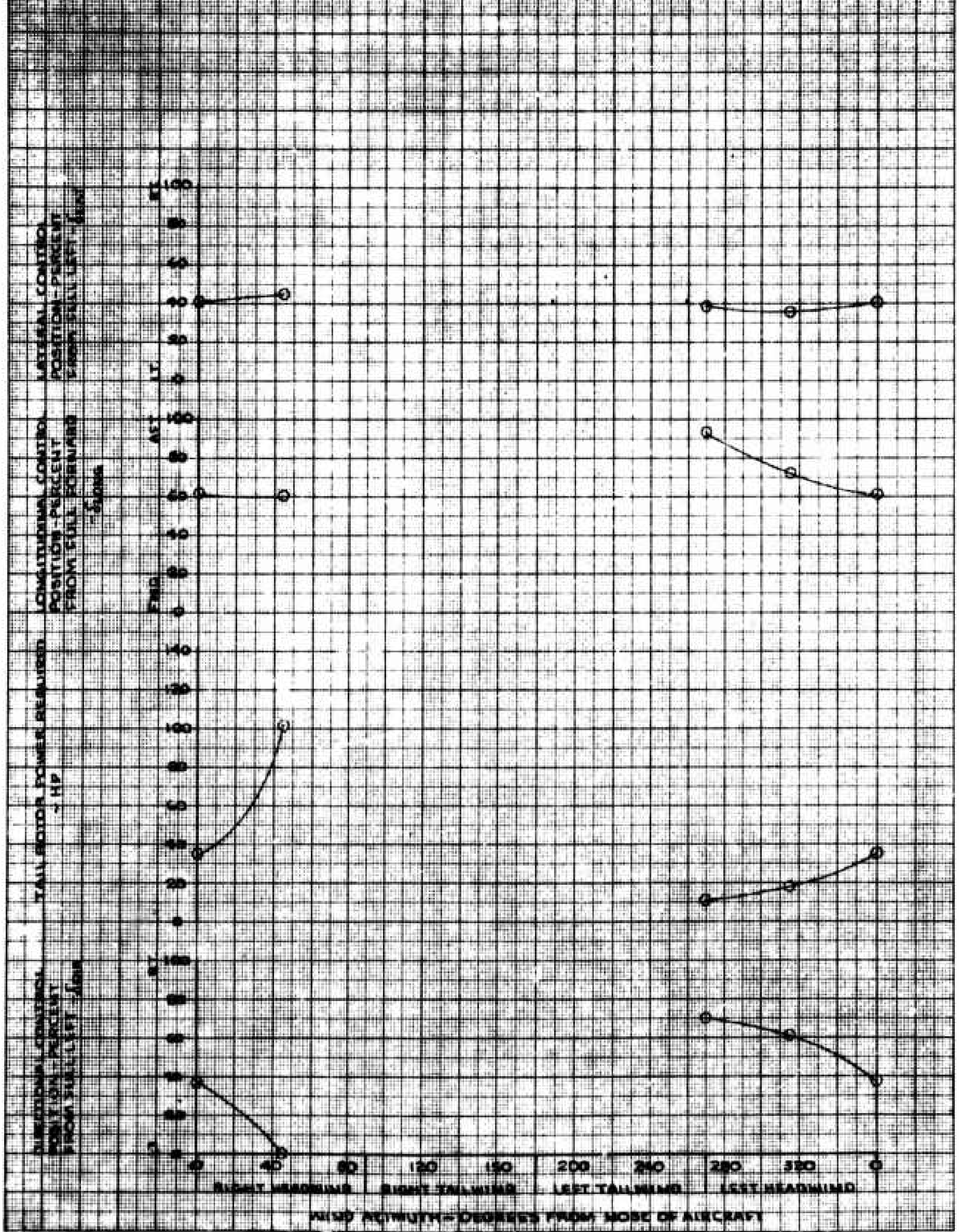
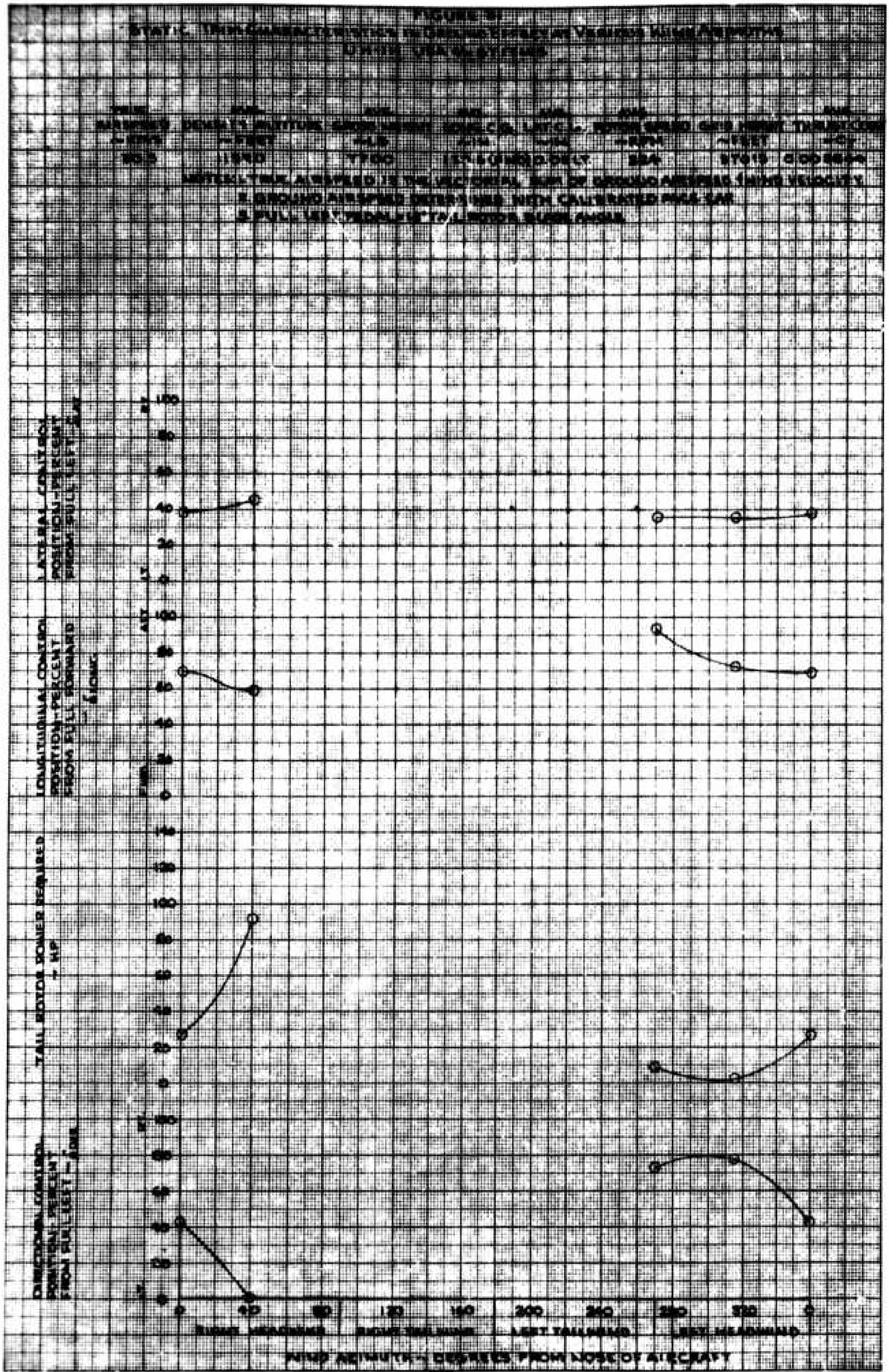

TAIL ROTOR POWER REQUIRED ~HP
LONGITUDINAL CONTROL POSITION~PERCENT FROM FULL FORWARD
LATERAL CONTROL POSITION~PERCENT FROM FULL LEFT
DIRECTIONAL CONTROL POSITION~PERCENT FROM FULL LEFT
0
40
80
120
160
200
240
280
320
0

FIGURE 30
STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS
UH-1H USA S/N [illegible]

| TRUE AIRSPEED ~KTAS | AVG. DENSITY ALTITUDE ~FEET | AVG. GROSS WEIGHT ~LB | AVG. LONG. C.G. ~IN. | AVG. LAT. C.G. ~IN. | AVG. ROTOR SPEED ~RPM | SKID HEIGHT ~FEET | AVG. THRUST COEF. ~$C_T$ |
|---|---|---|---|---|---|---|---|
| [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | 324 | [illegible] | [illegible] |

NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = [illegible] TAIL ROTOR BLADE ANGLE




LATERAL CONTROL POSITION-PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION-PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION-PERCENT FROM FULL LEFT
0
40
80
120
160
200
240
280
320
0
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

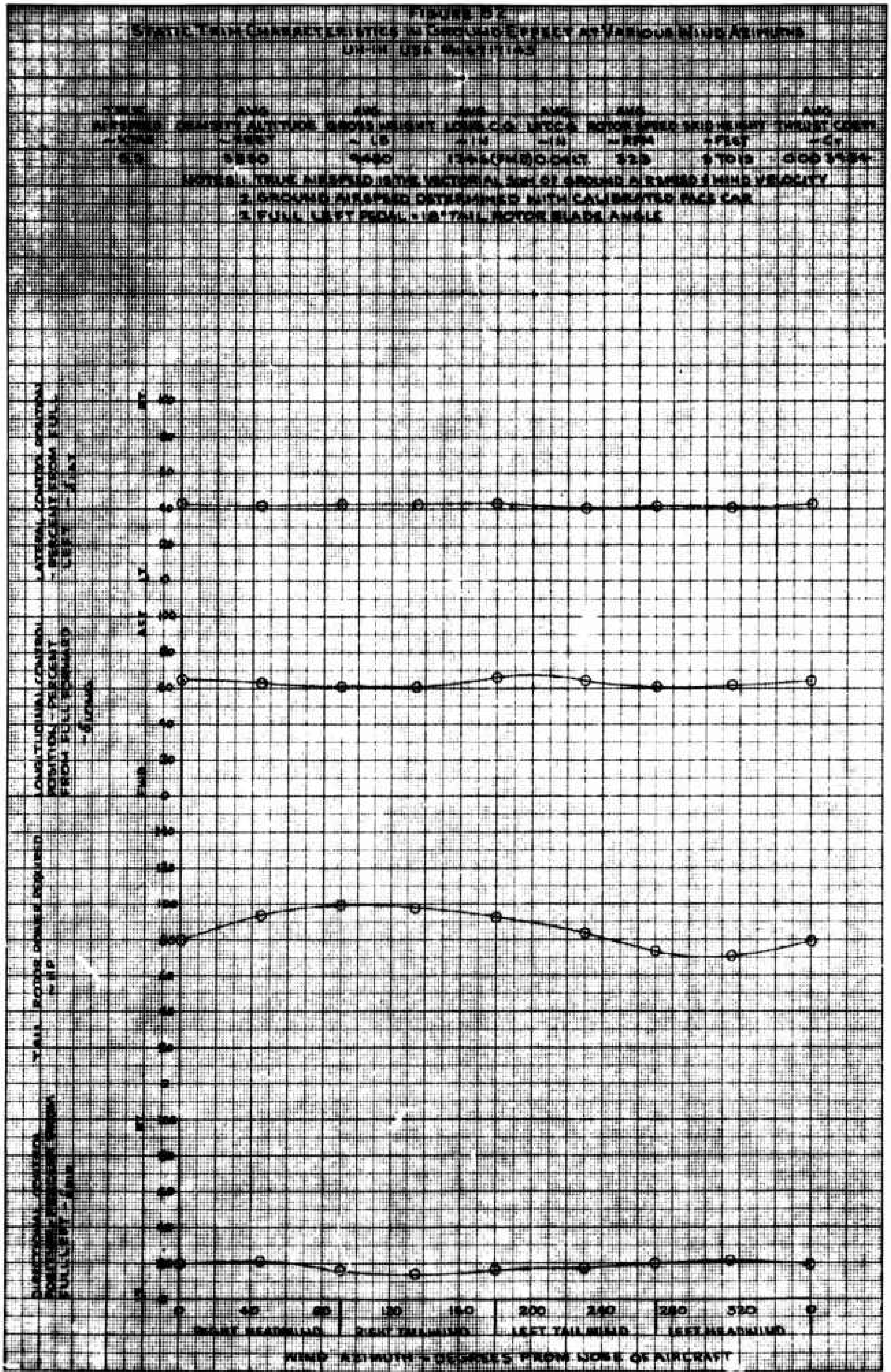
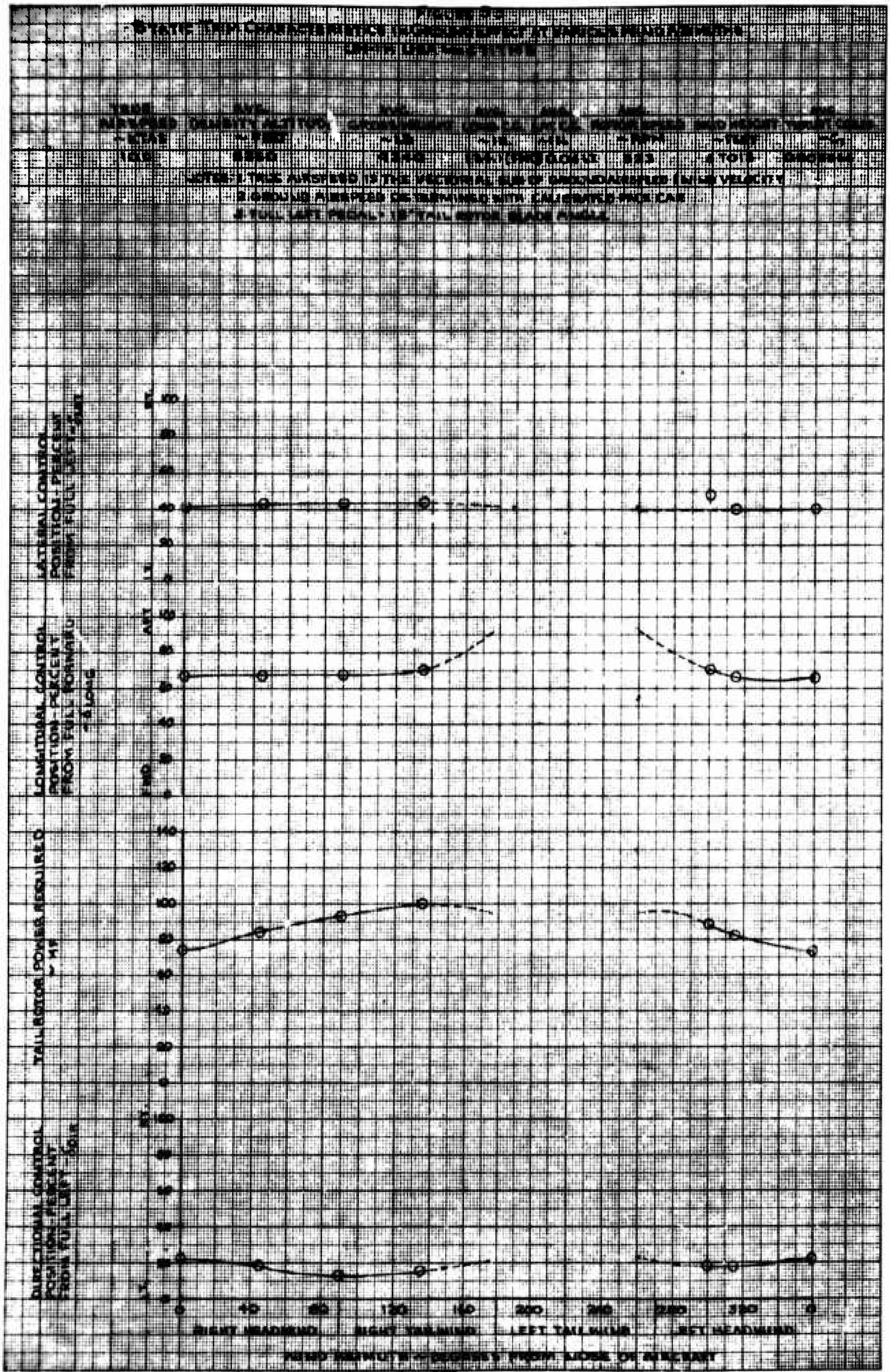
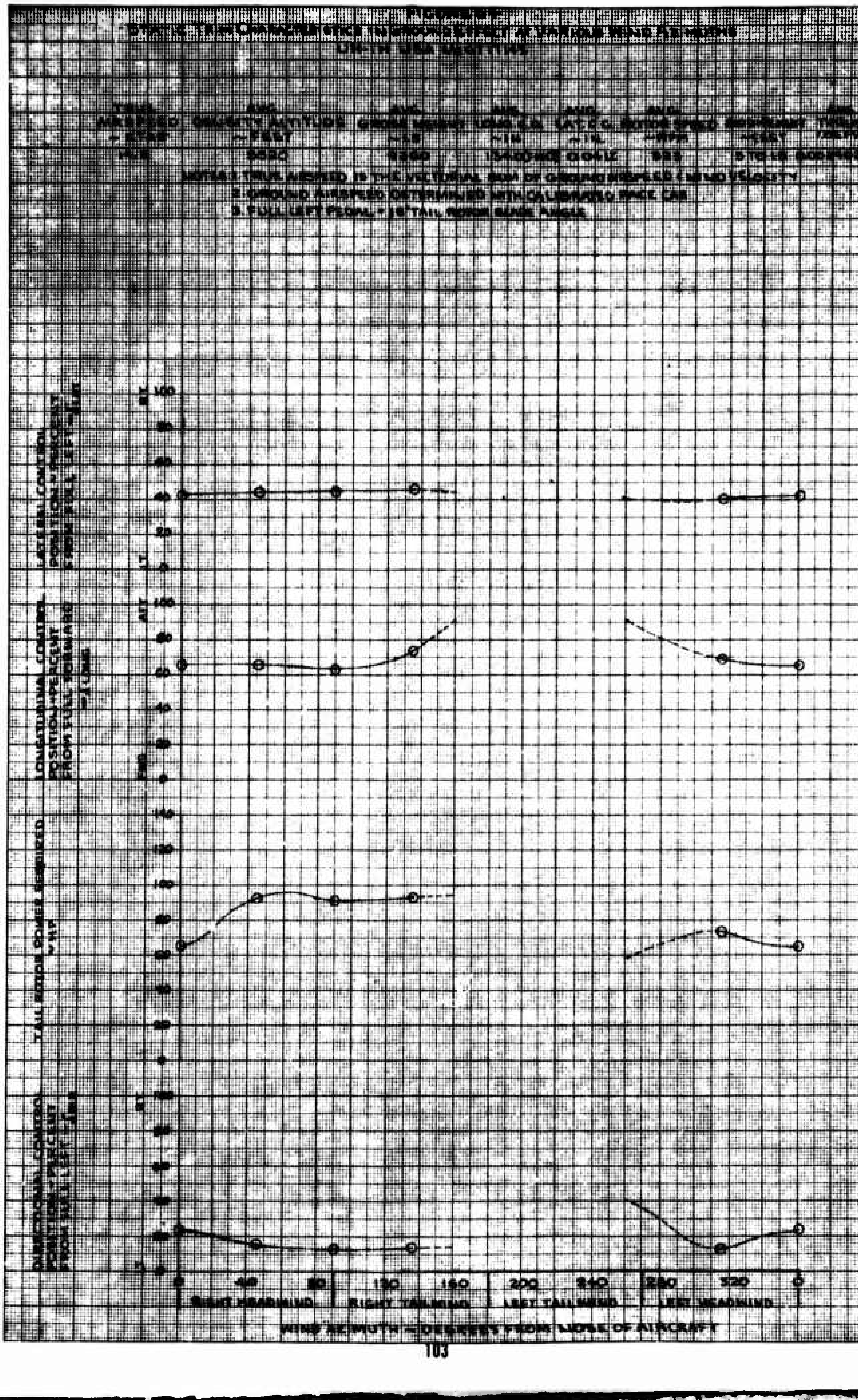
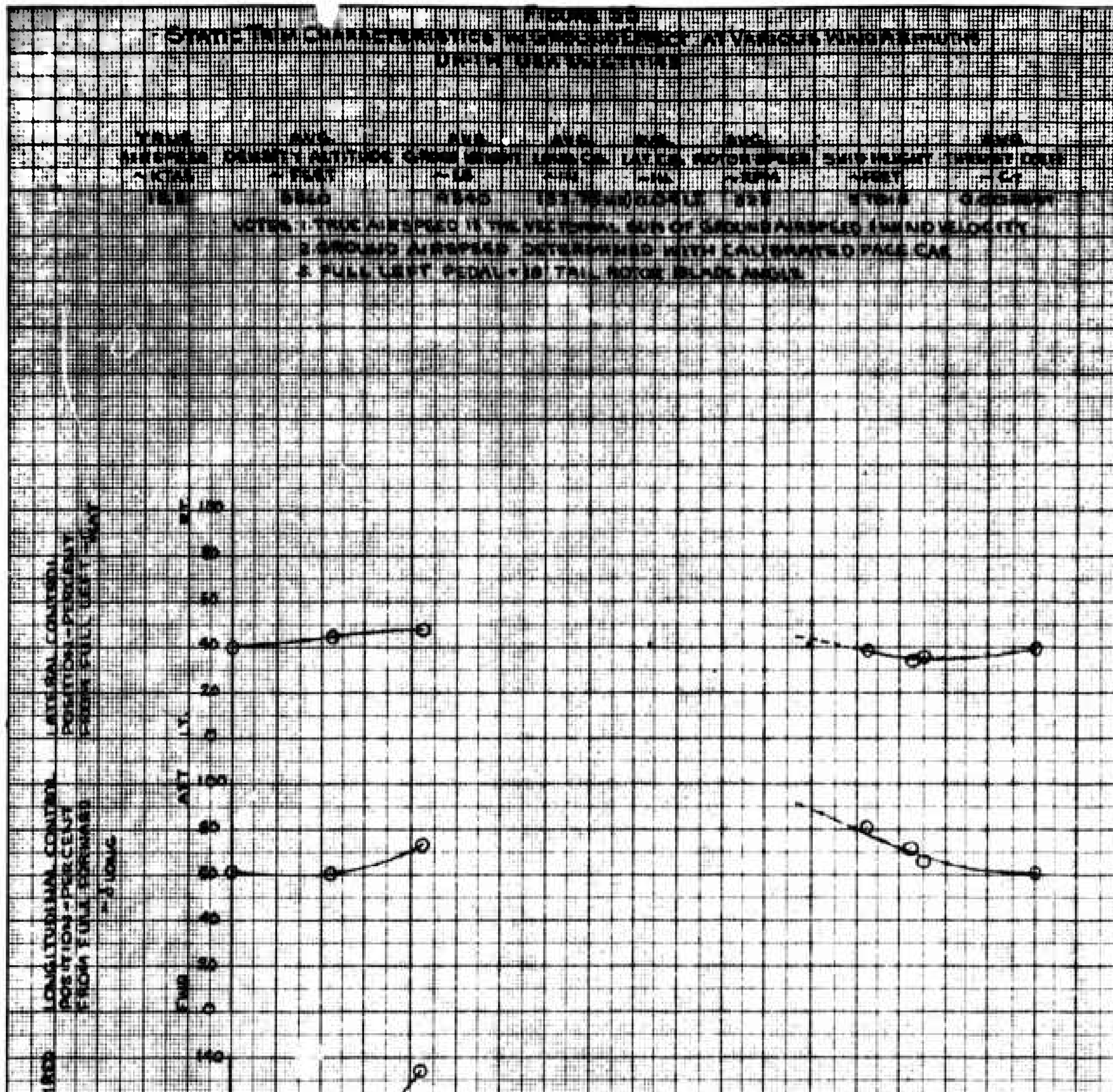
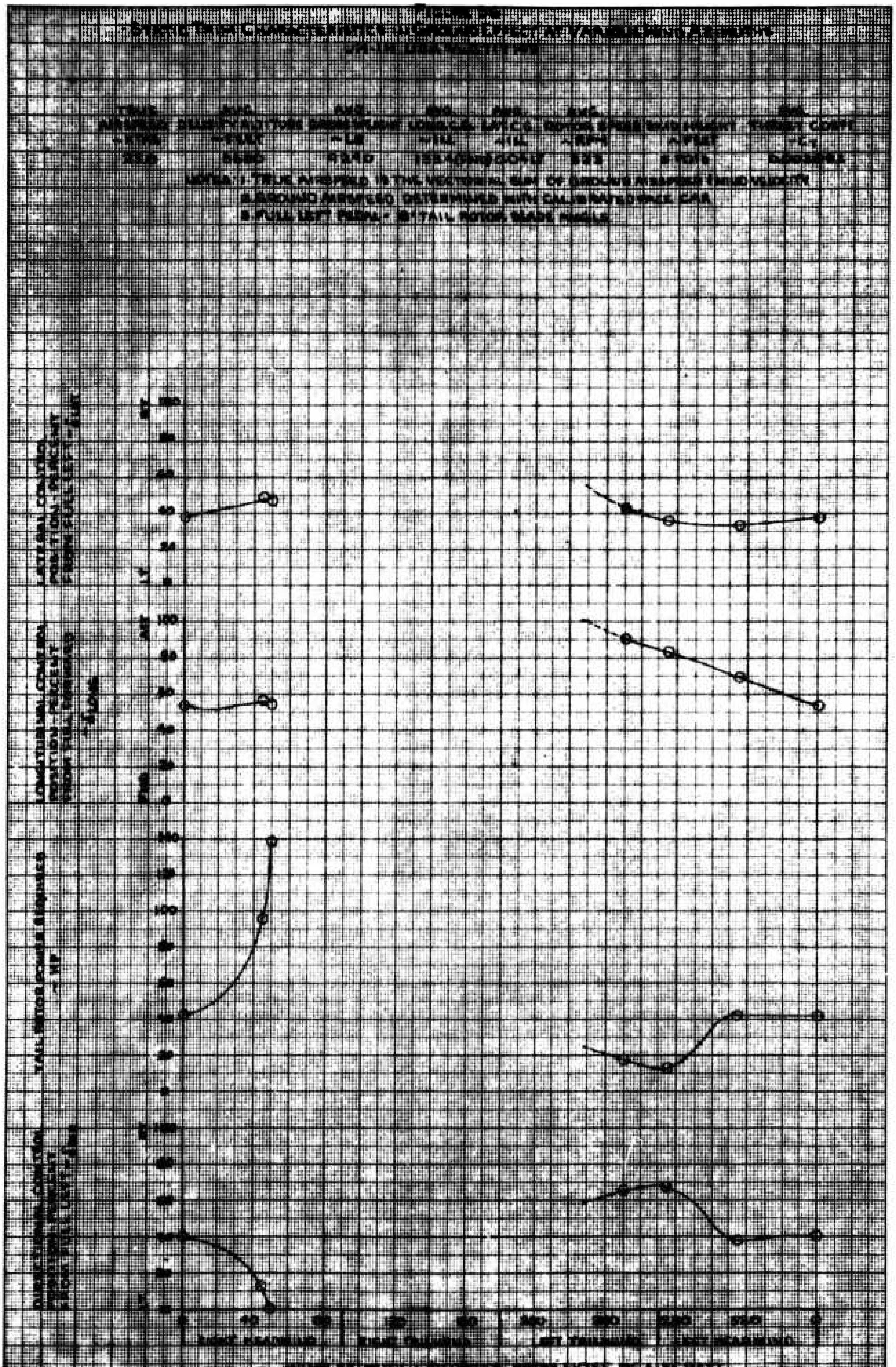


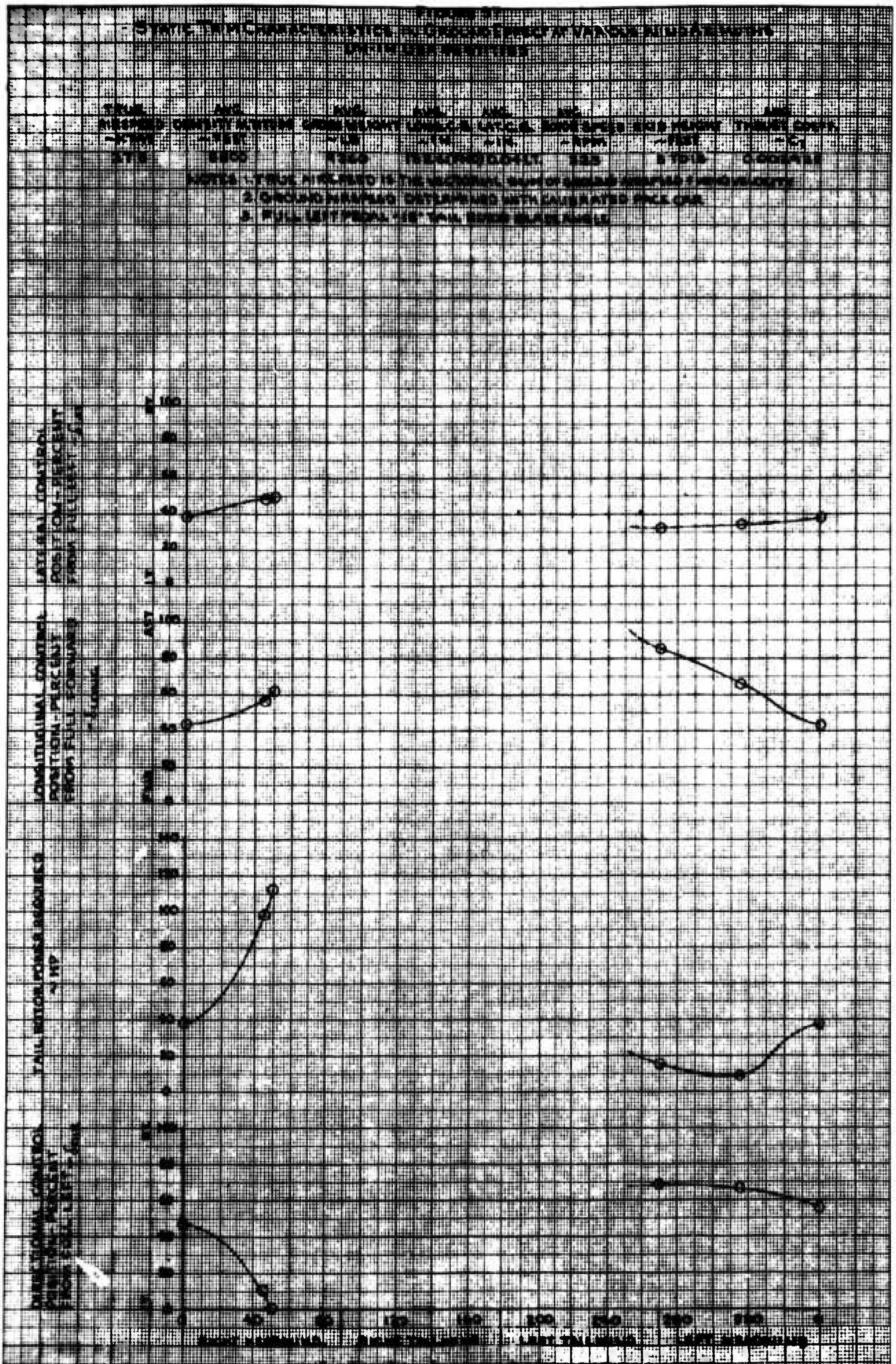
FIGURE 52
STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED & WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = 18° TAIL ROTOR BLADE ANGLE
LATERAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

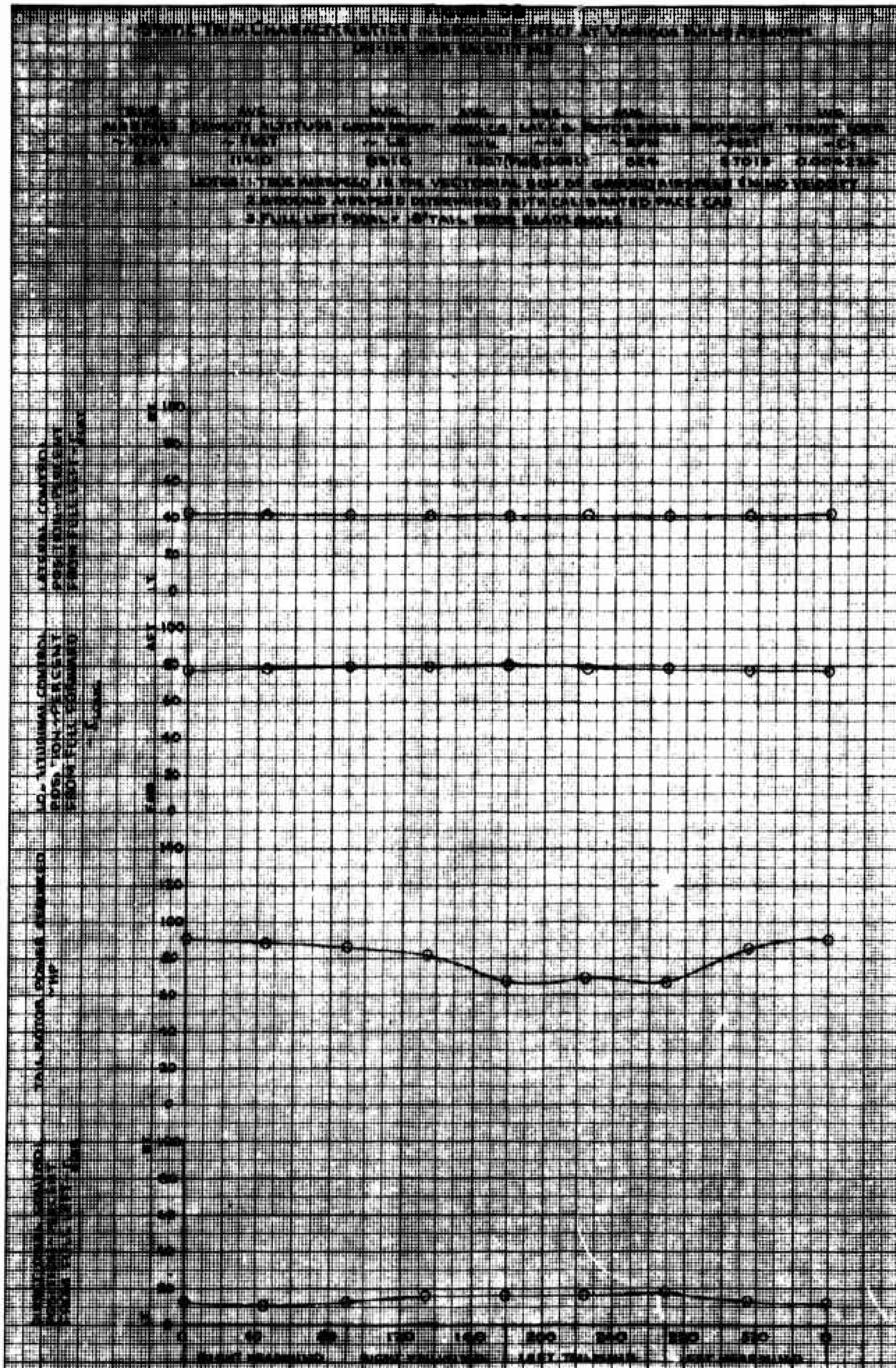

GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
LATERAL CONTROL POSITION-PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION-PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION-PERCENT FROM FULL LEFT
RT.
LT.
AFT
FWD
0
40
80
120
160
200
240
280
320
0
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

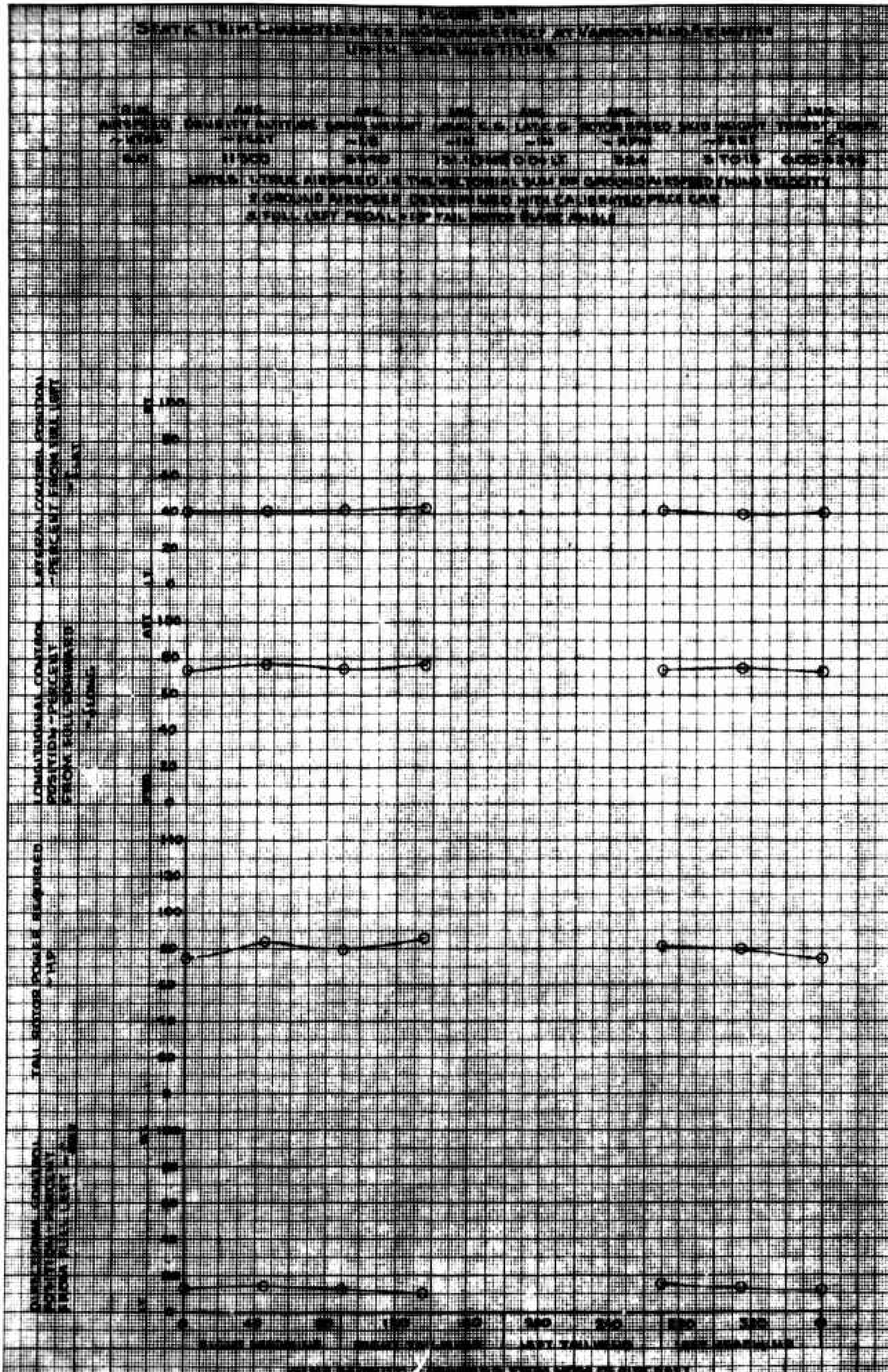

NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED & WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = 18° TAIL ROTOR BLADE ANGLE
LATERAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
0
40
80
120
160
200
240
280
320
0
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

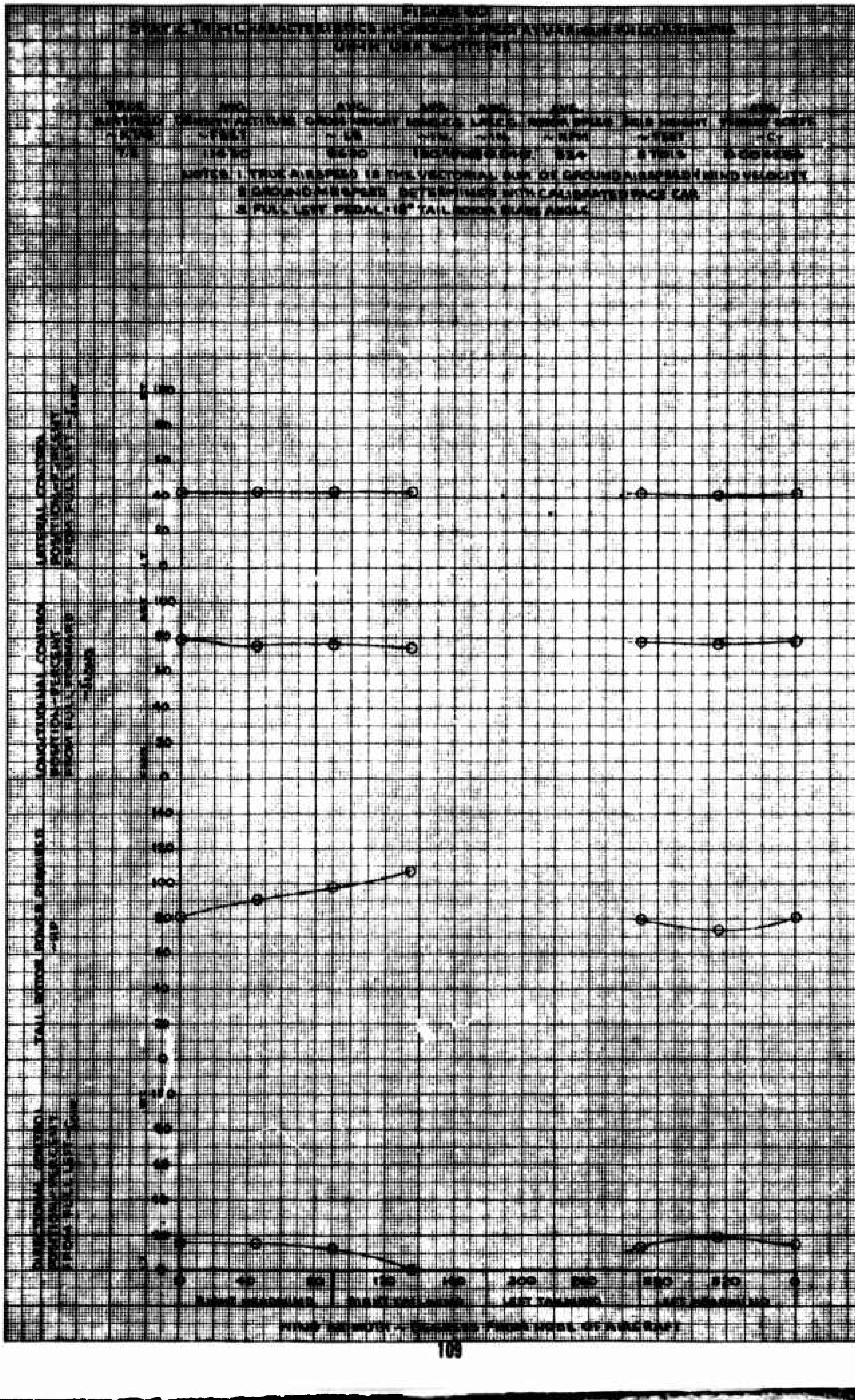

NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED & WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
LATERAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

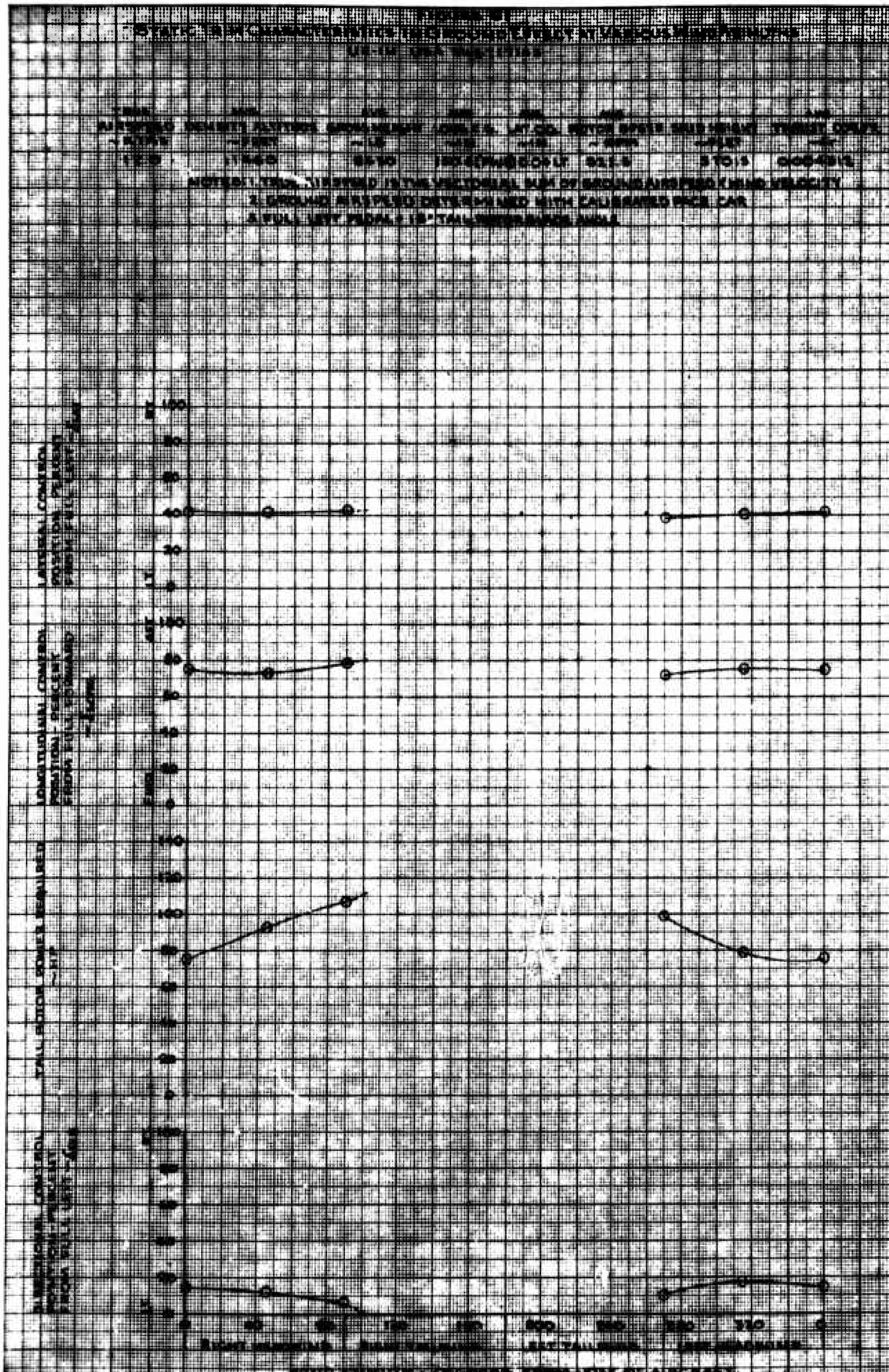



2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
LATERAL CONTROL POSITION - PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION - PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION - PERCENT FROM FULL LEFT






NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = 15° TAIL ROTOR BLADE ANGLE



LATERAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED ~ HP
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

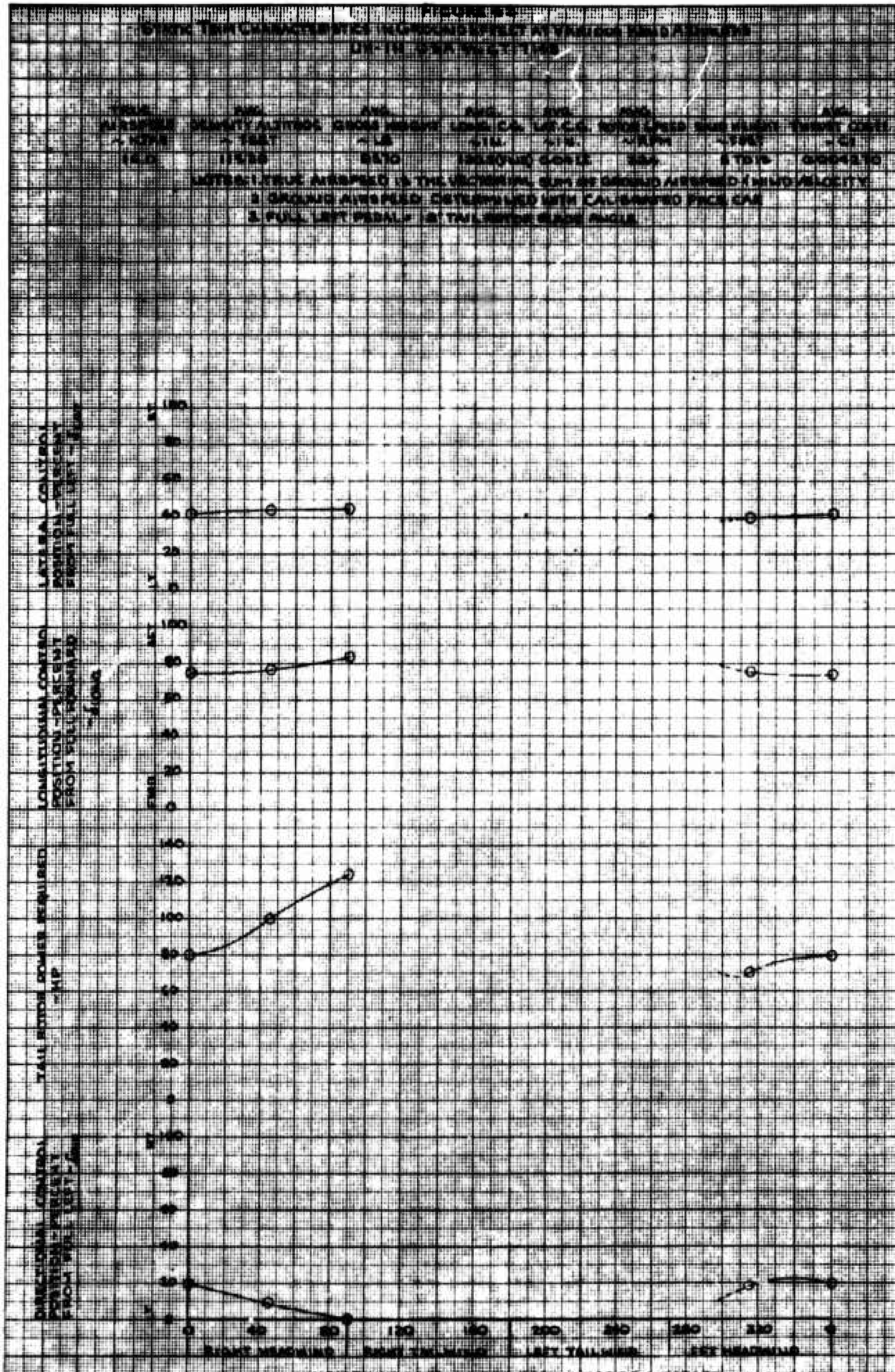


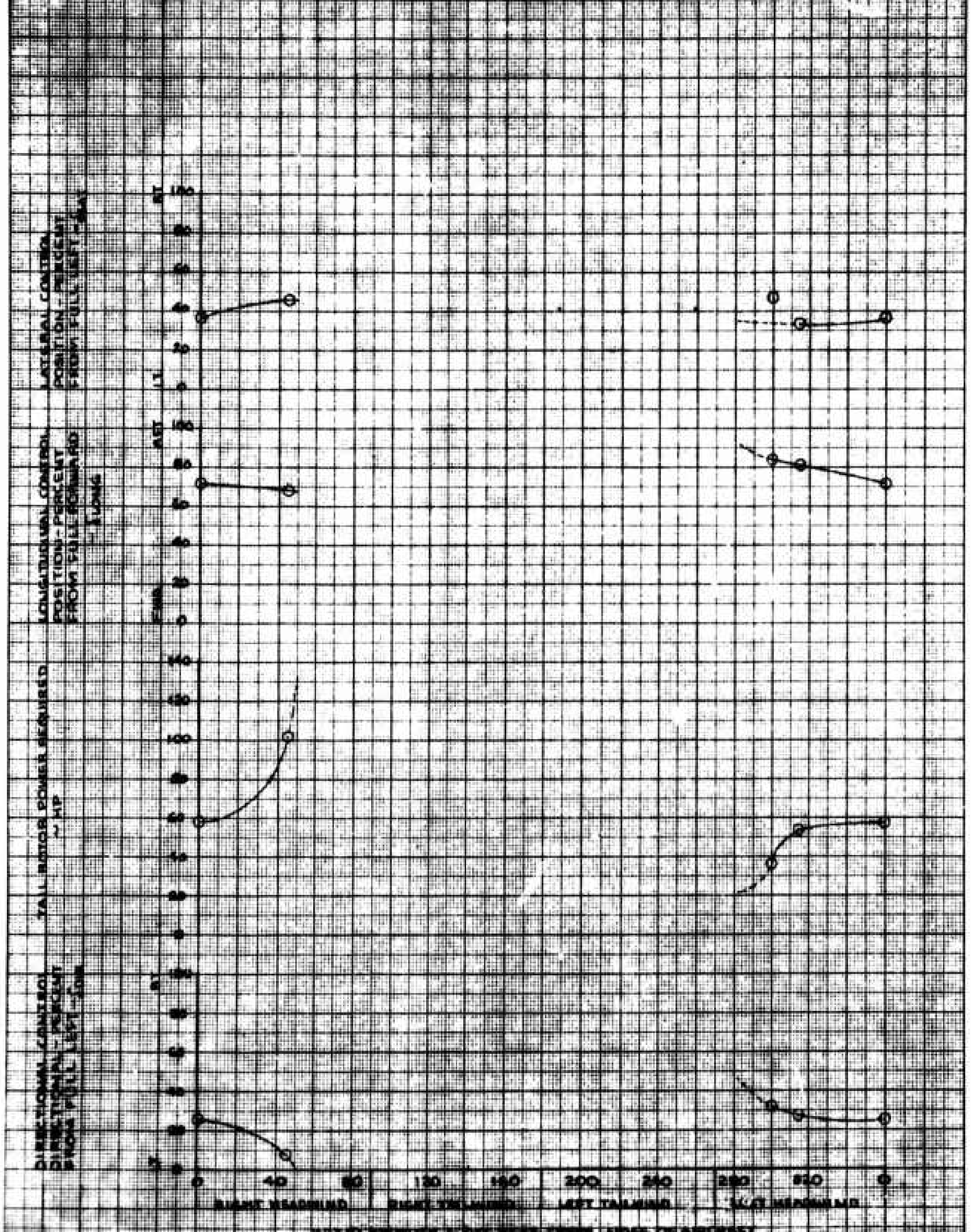
FIGURE 39

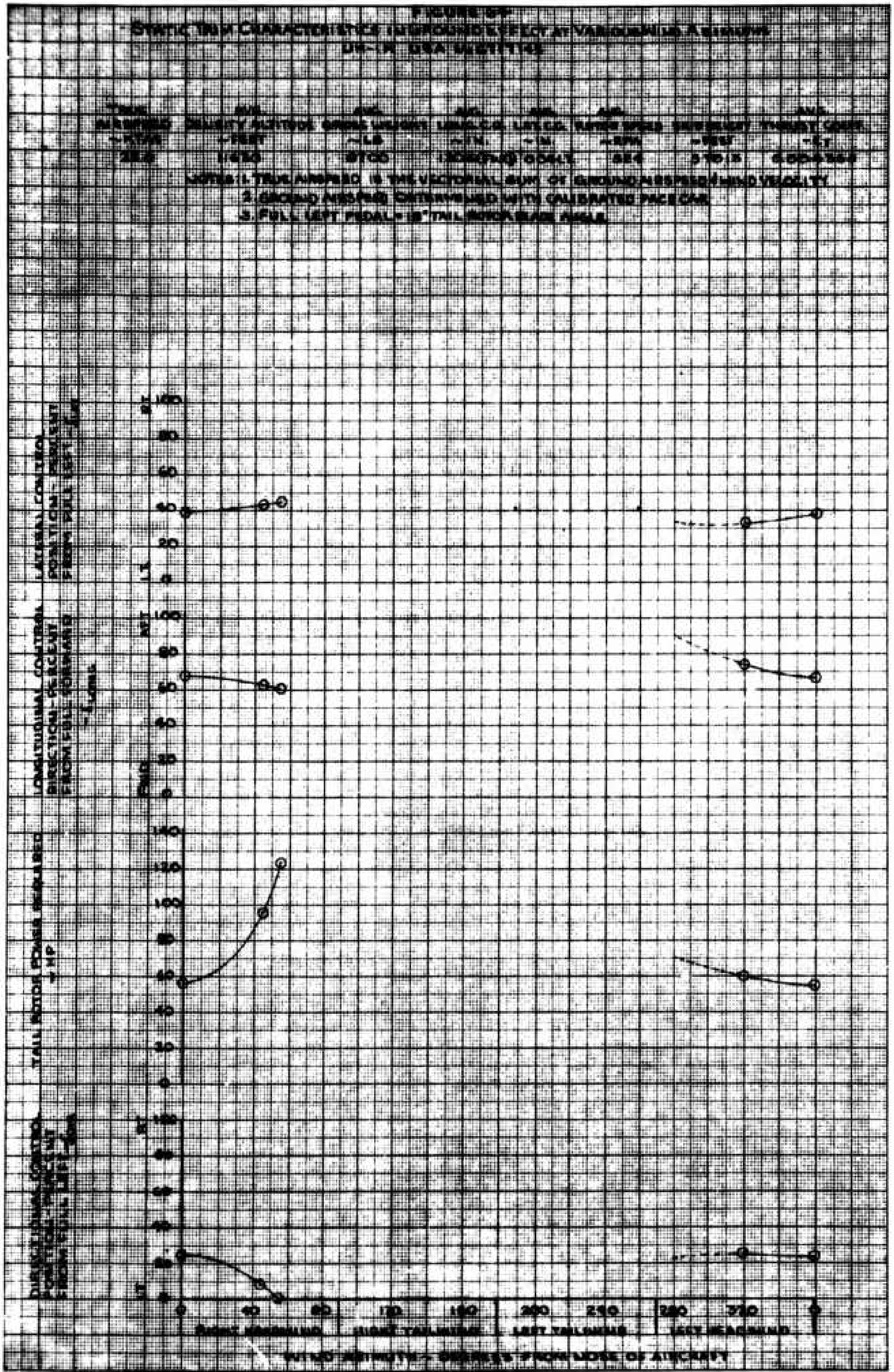
STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS

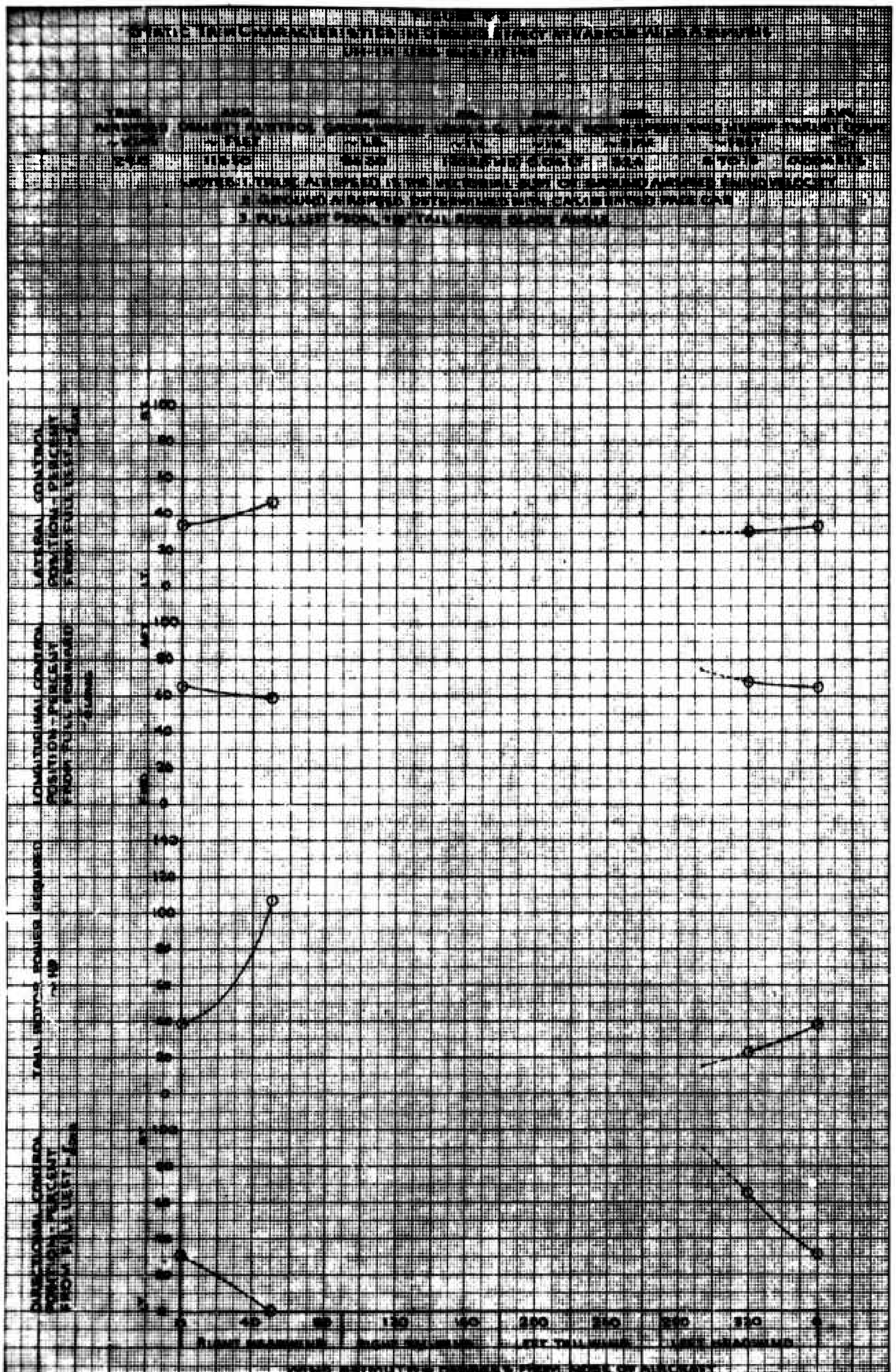
UH-1H USA S/N [illegible]

| TRUE AIRSPEED ~KNOTS | AVG. DENSITY ALTITUDE ~FEET | AVG. GROSS WEIGHT ~LB | AVG. LONG. C.G. ~IN | AVG. LAT. C.G. ~IN | AVG. ROTOR SPEED ~RPM | SKID HEIGHT ~FEET | AVG. THRUST COEFF ~$C_T$ |
|---|---|---|---|---|---|---|---|
| 20.5 | 11500 | 8440 | 130.8 (MID) | 0.04 LT | 324 | 4 TO 15 | 0.004283 |

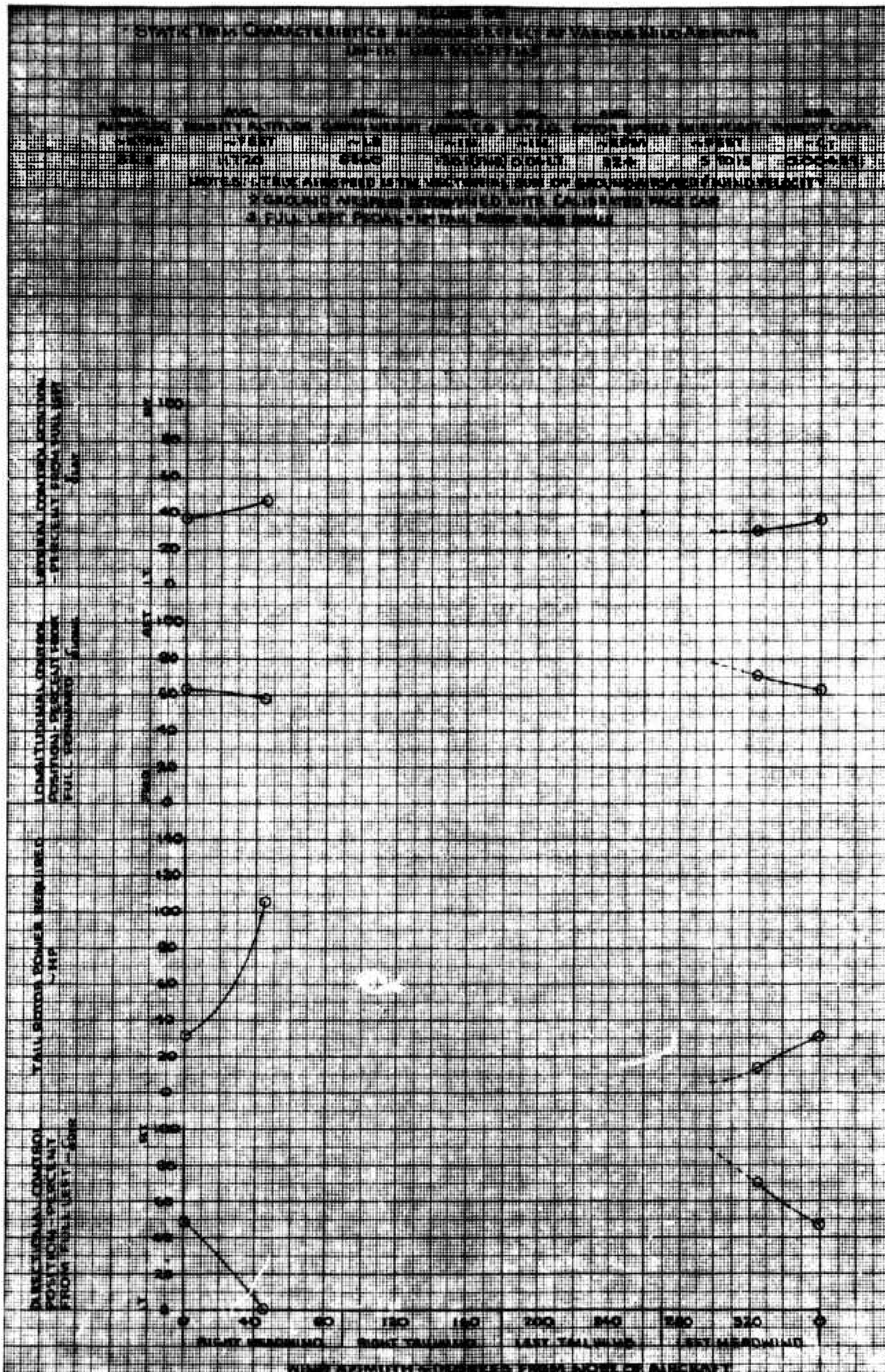
NOTES: 1. TRUE AIRSPEED IS THE VECTORIAL SUM OF GROUND AIRSPEED AND WIND VELOCITY
2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = 18° TAIL ROTOR BLADE ANGLE




STATIC TRIM CHARACTERISTICS IN GROUND EFFECT AT VARIOUS WIND AZIMUTHS
TAIL ROTOR POWER REQUIRED ~ HP
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT



0
40
80
120
160
200
240
280
320
0


TAIL ROTOR POWER REQUIRED ~HP
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
WIND AZIMUTH ~ DEGREES FROM NOSE OF AIRCRAFT

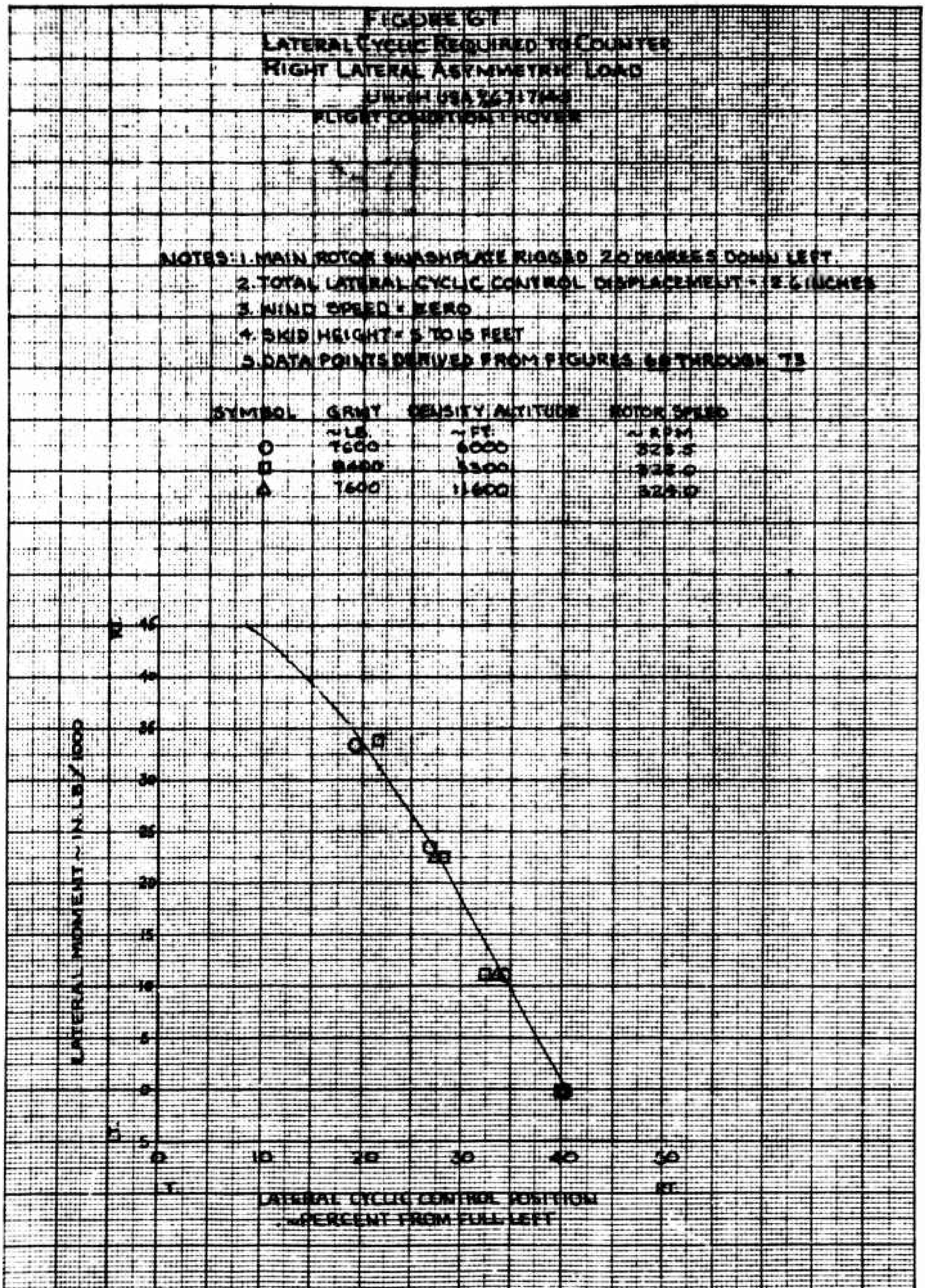
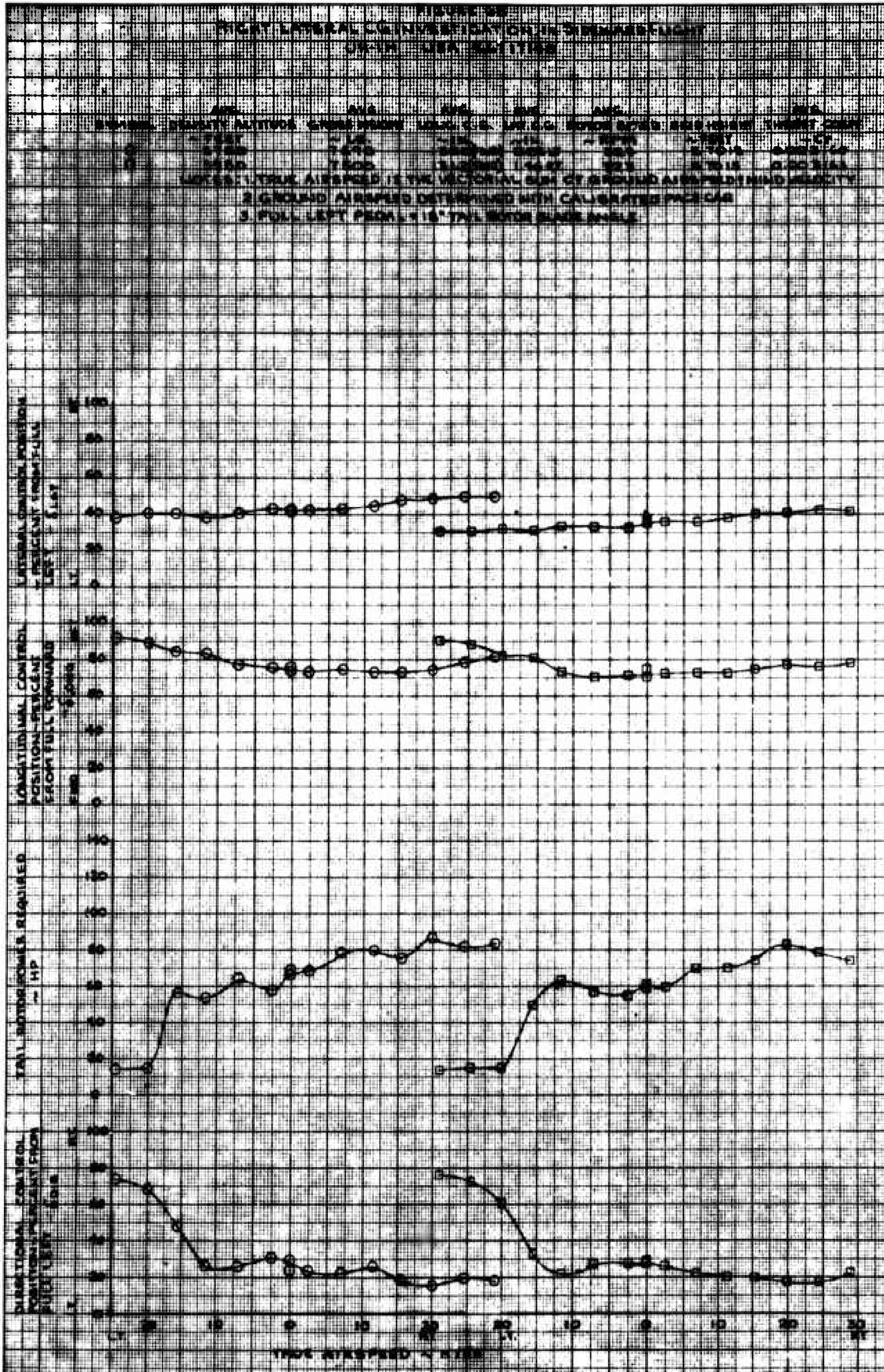
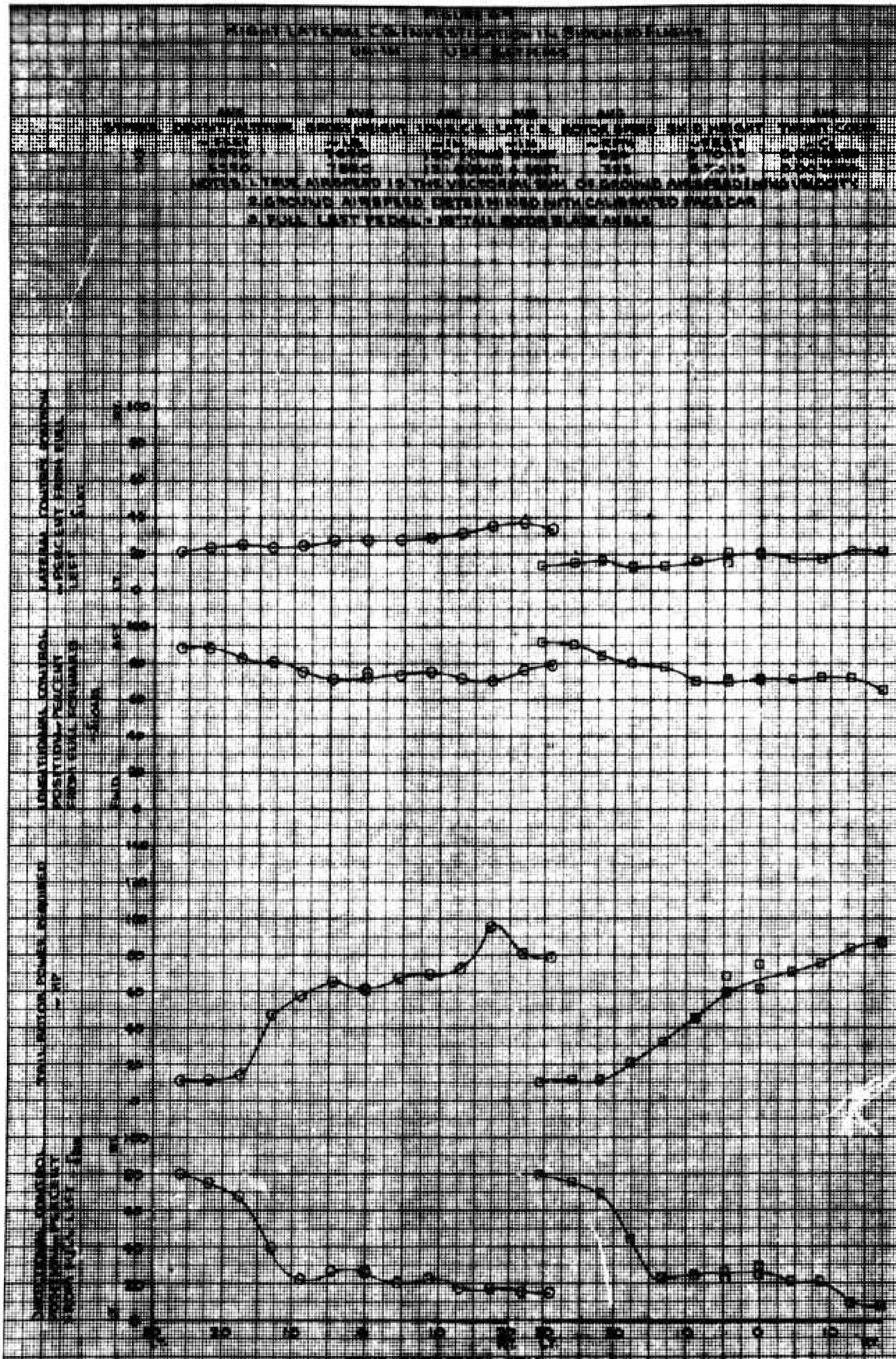
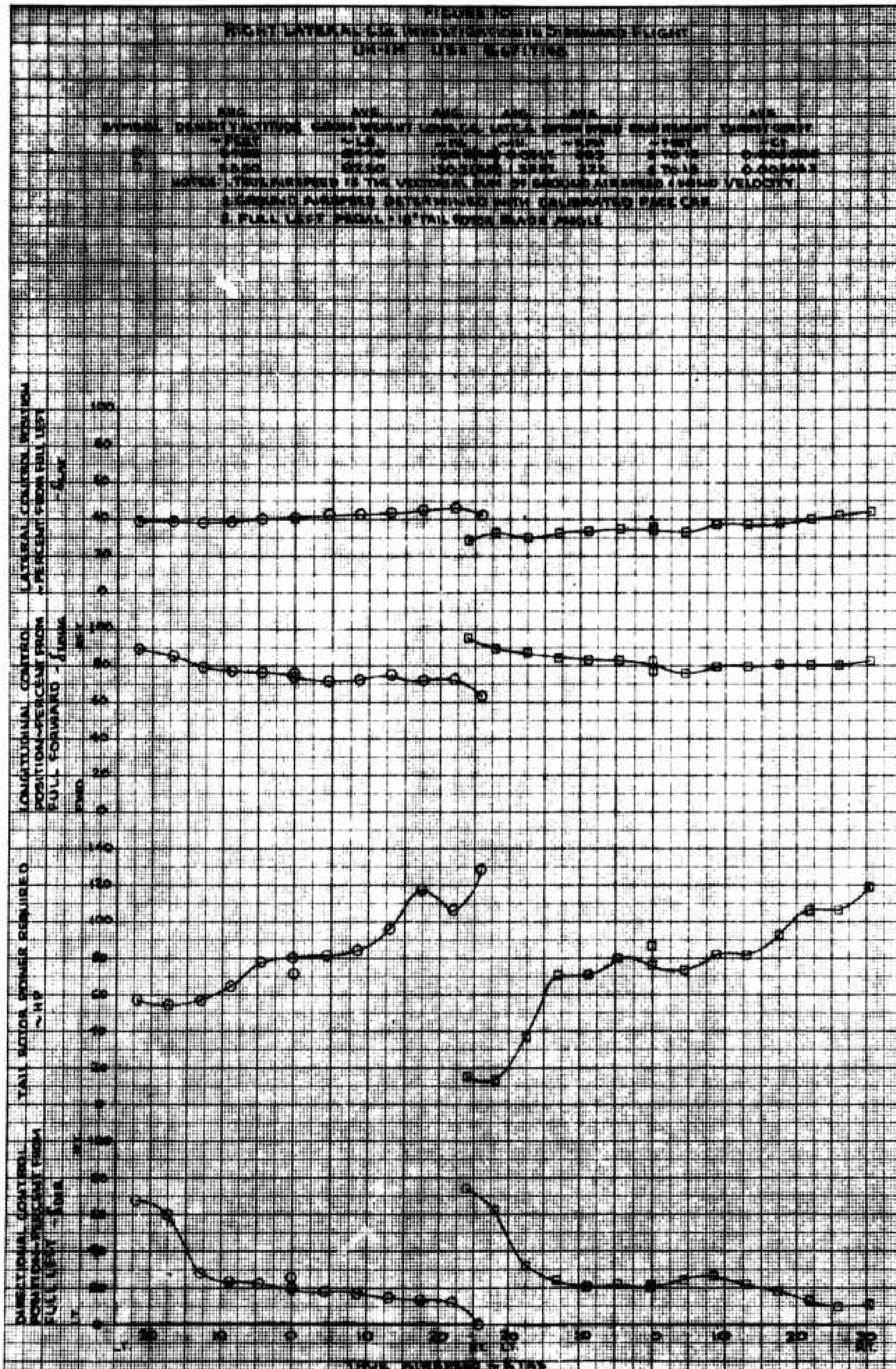


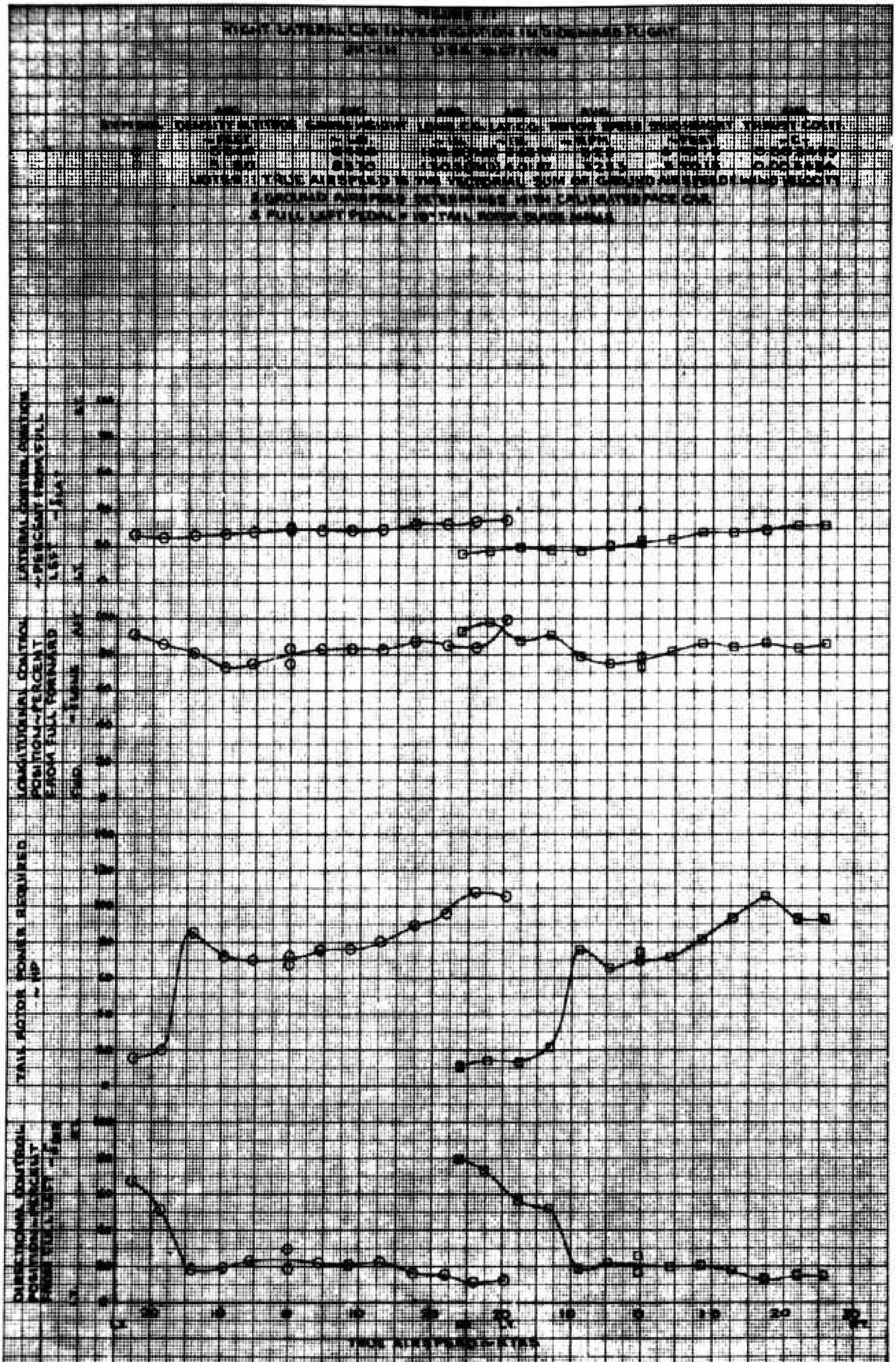
FIGURE 67
LATERAL CYCLIC REQUIRED TO COUNTER
RIGHT LATERAL ASYMMETRIC LOAD
FLIGHT CONDITION: HOVER
NOTES: 1. MAIN ROTOR SWASHPLATE RIGGED 2.0 DEGREES DOWN LEFT.
2. TOTAL LATERAL CYCLIC CONTROL DISPLACEMENT = 12.6 INCHES
3. WIND SPEED = ZERO
4. SKID HEIGHT = 5 TO 15 FEET
5. DATA POINTS DERIVED FROM FIGURES 68 THROUGH 73
SYMBOL GRWT DENSITY ALTITUDE ROTOR SPEED
~LB. ~FT. ~RPM
○ 7600 6000 323.5
□ 8400 5300
△ 7600 11600 324.0
LATERAL MOMENT ~ IN.LB./1000
LT.
RT.
LATERAL CYCLIC CONTROL POSITION
~PERCENT FROM FULL LEFT

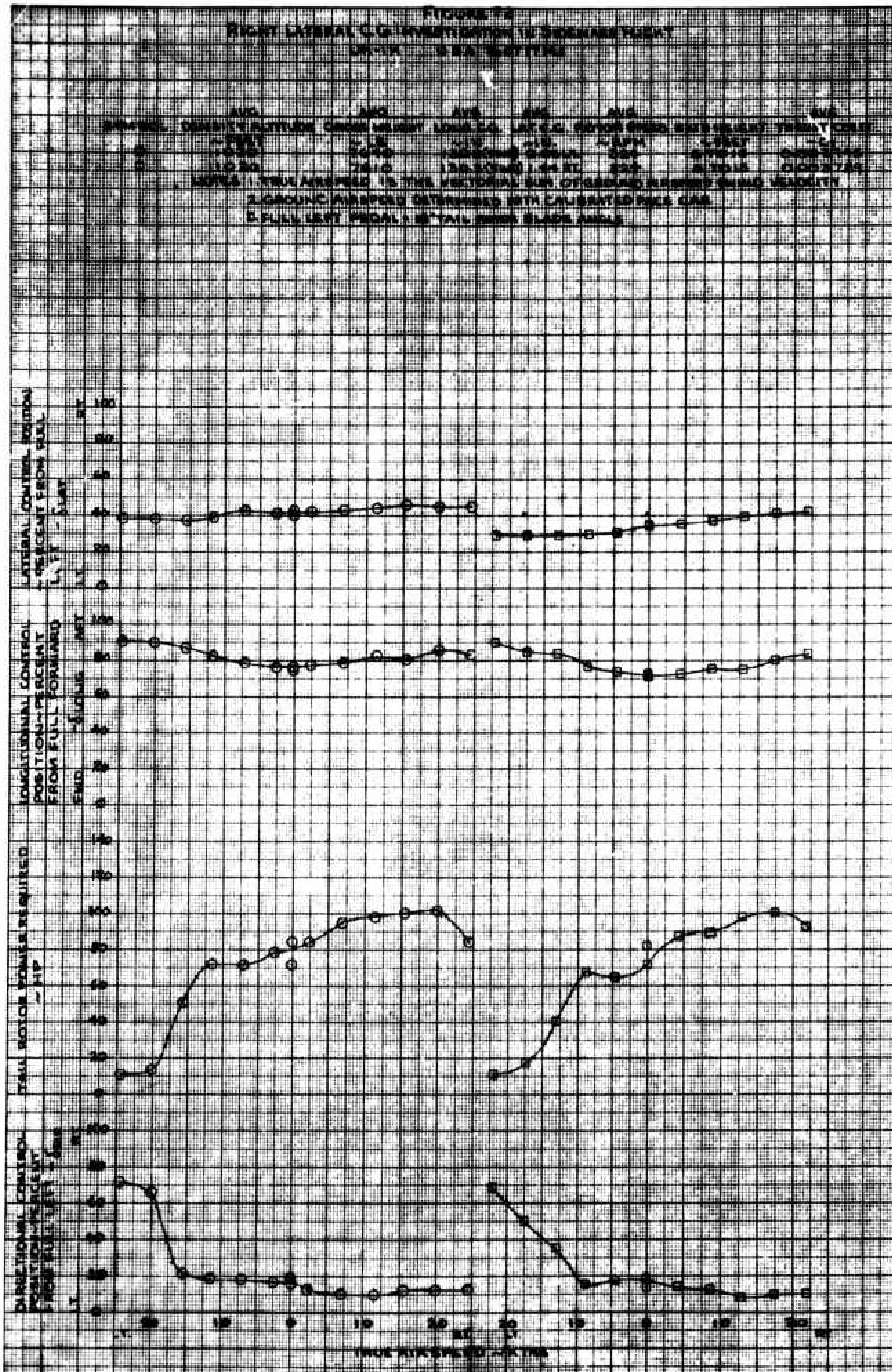

2. GROUND AIRSPEED DETERMINED WITH CALIBRATED PACE CAR
3. FULL LEFT PEDAL = +15° TAIL ROTOR BLADE ANGLE
TAIL ROTOR POWER REQUIRED ~ HP
TRUE AIRSPEED ~ KTAS
20
10
0
10
20
30



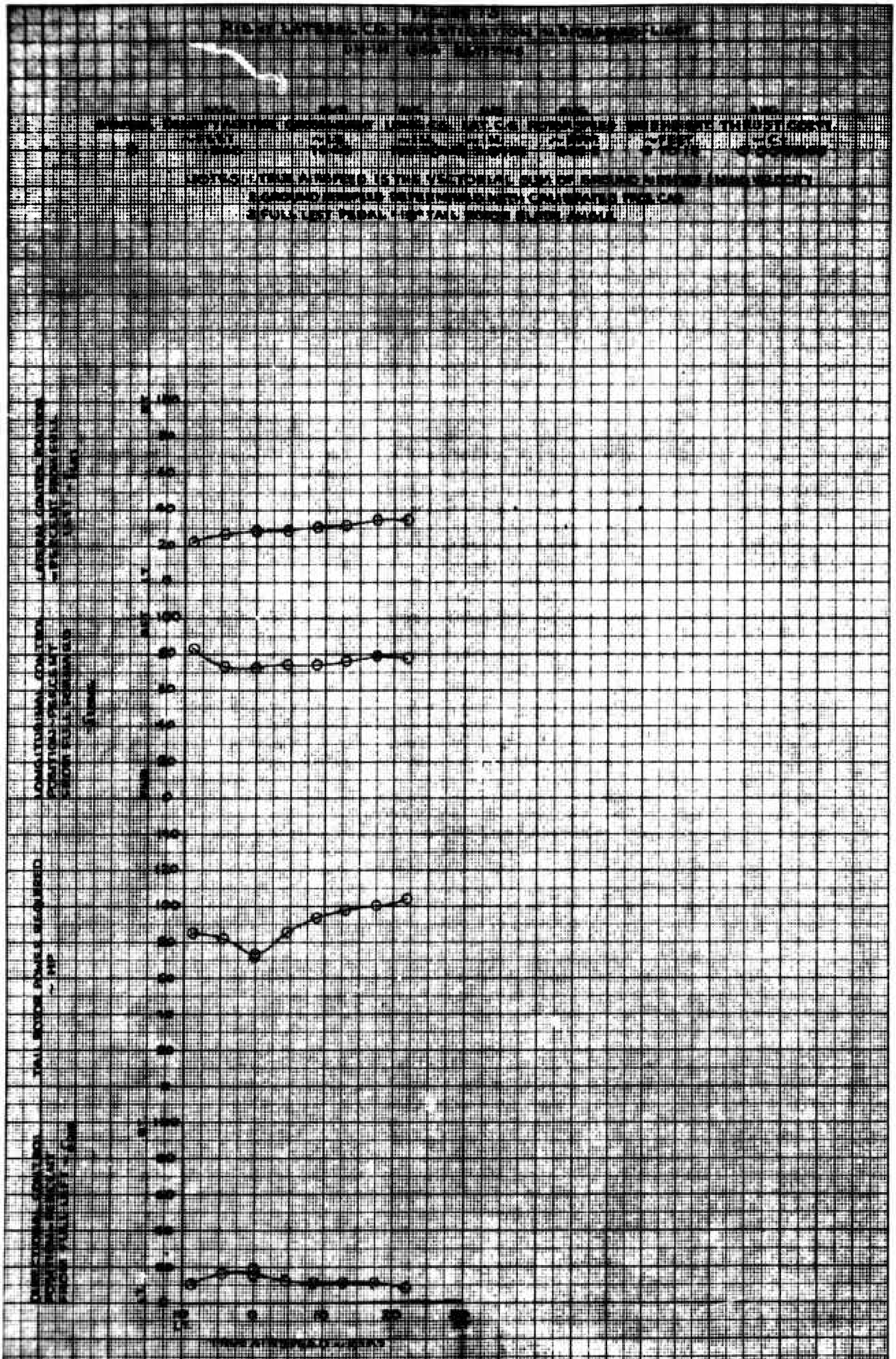


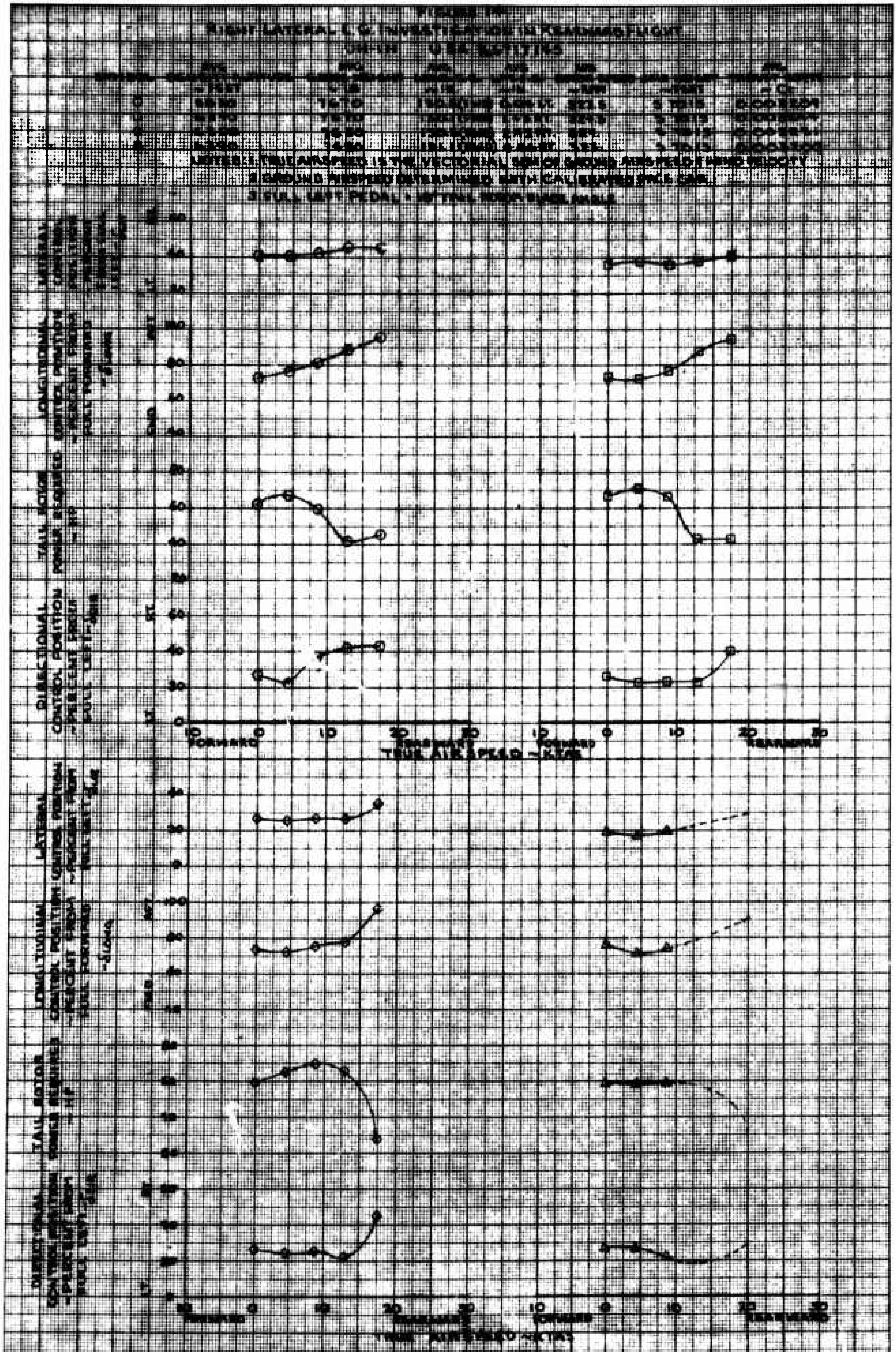
UH-1H
TRUE AIRSPEED ~ KTAS





RIGHT LATERAL C.G. INVESTIGATION IN SIDEWARD FLIGHT
TAIL ROTOR POWER REQUIRED ~ HP
LONGITUDINAL CONTROL POSITION ~ PERCENT FROM FULL FORWARD
LATERAL CONTROL POSITION ~ PERCENT FROM FULL
DIRECTIONAL CONTROL POSITION ~ PERCENT FROM FULL LEFT
TRUE AIRSPEED ~ KNOTS




RIGHT LATERAL C.G. INVESTIGATION IN REARWARD FLIGHT
UH-1H
FORWARD
REARWARD
TRUE AIRSPEED ~ KTAS

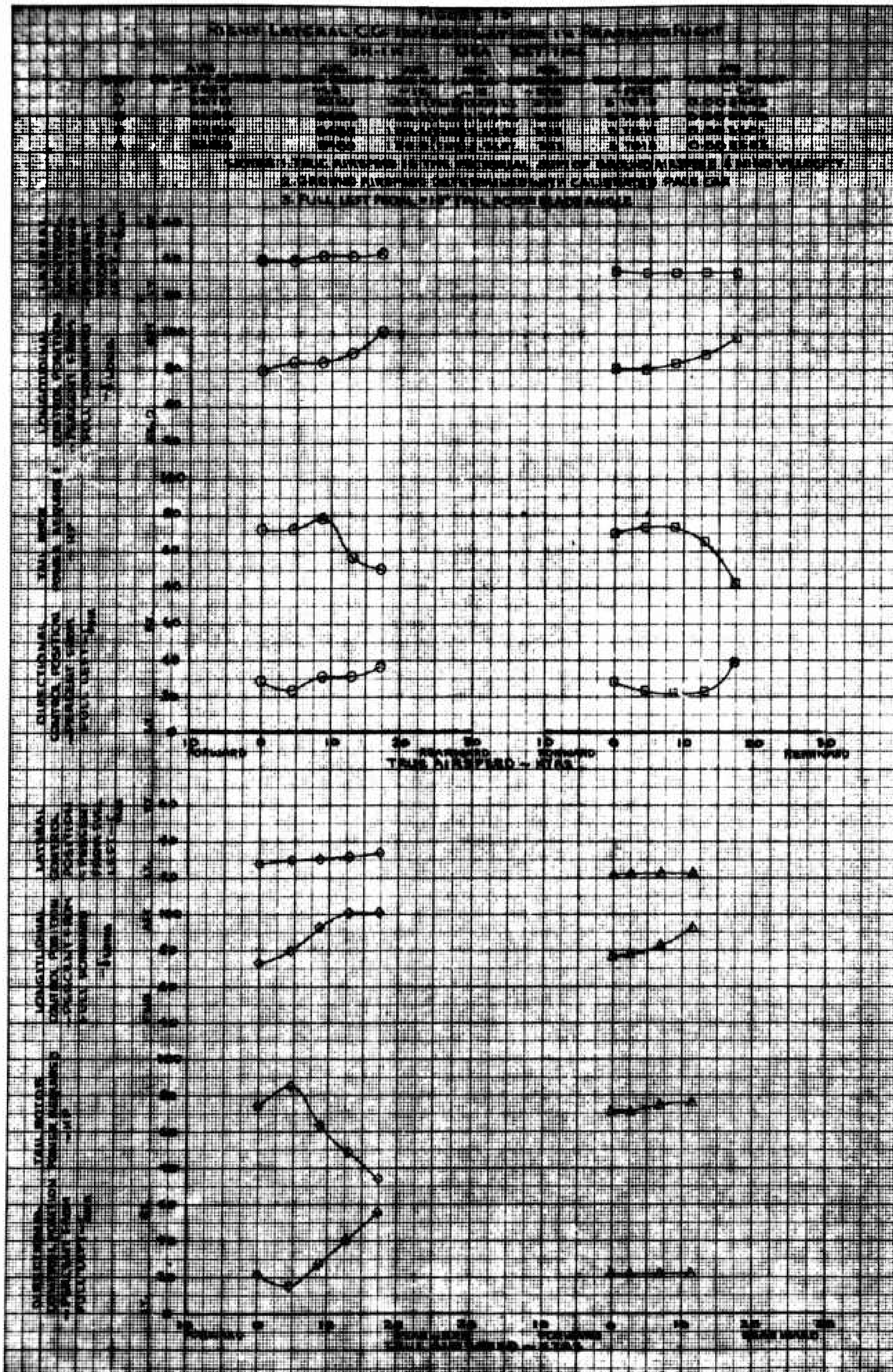


FIGURE 15
TRUE AIRSPEED ~ KTS
FORWARD
REARWARD

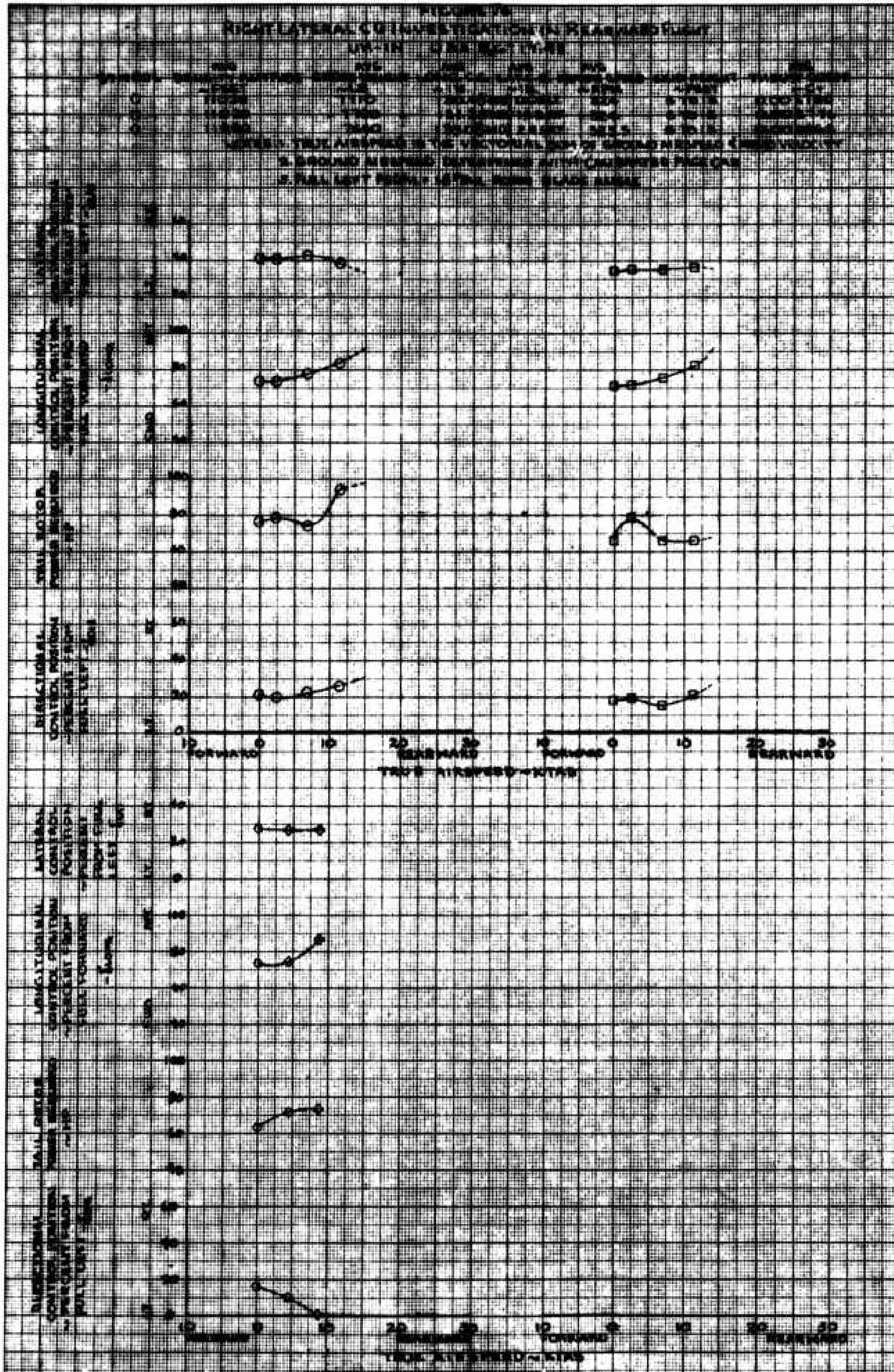
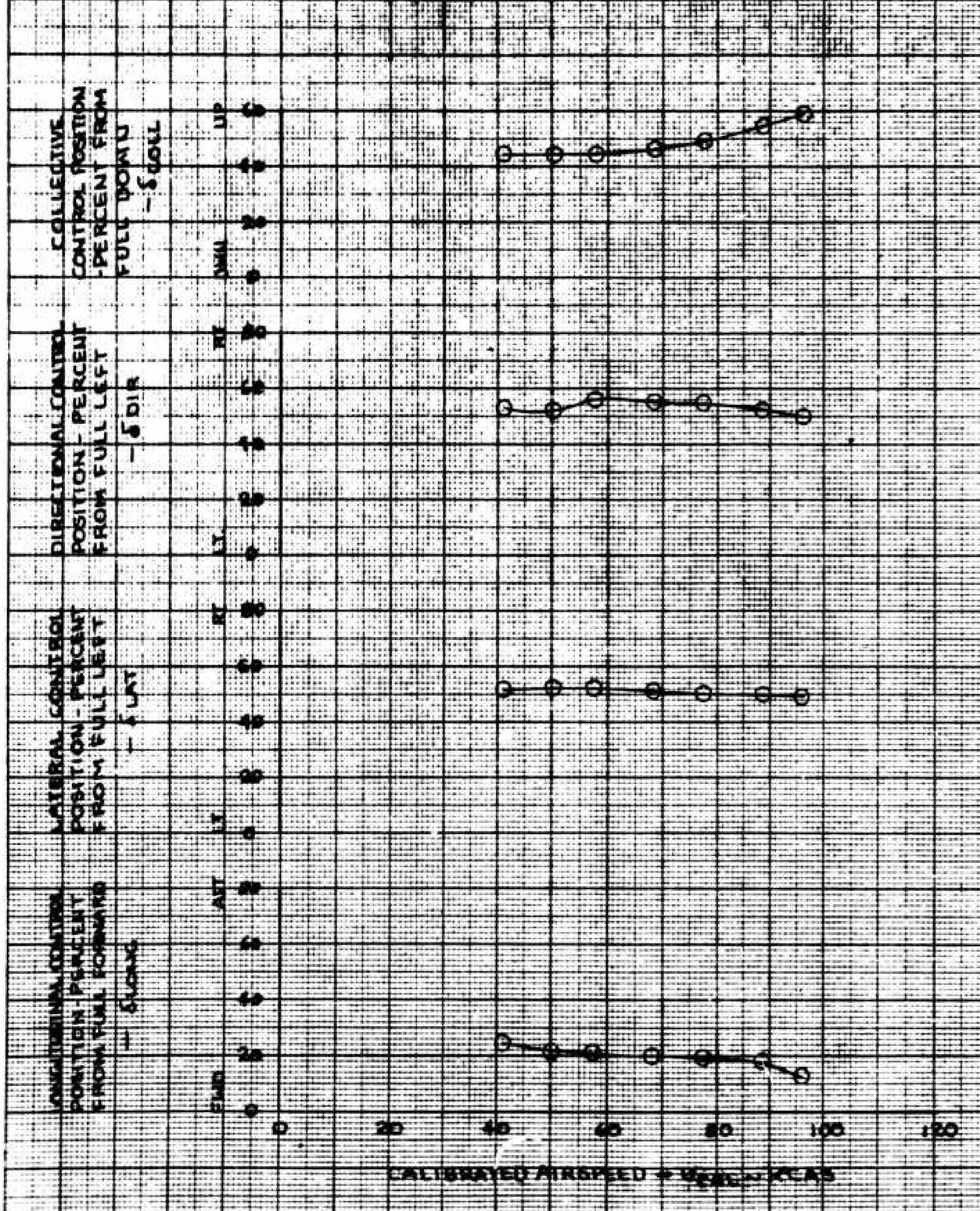


FIGURE 77
STATIC TRIM CHARACTERISTICS
UH-1H USA S/N 67-17145
FIRE SUPPRESSION KIT INSTALLED

| DENSITY ALTITUDE ~FEET | GROSS WEIGHT ~LB | LONGITUDINAL C.G. ~IN | LATERAL C.G. ~IN | ROTOR SPEED ~RPM | FLIGHT CONDITION | THRUST COEFF ~CT |
|---|---|---|---|---|---|---|
| 6340 | 8520 | 152 (AFT) | 0.3 LT | 322 | LVL FLT | 0.003622 |



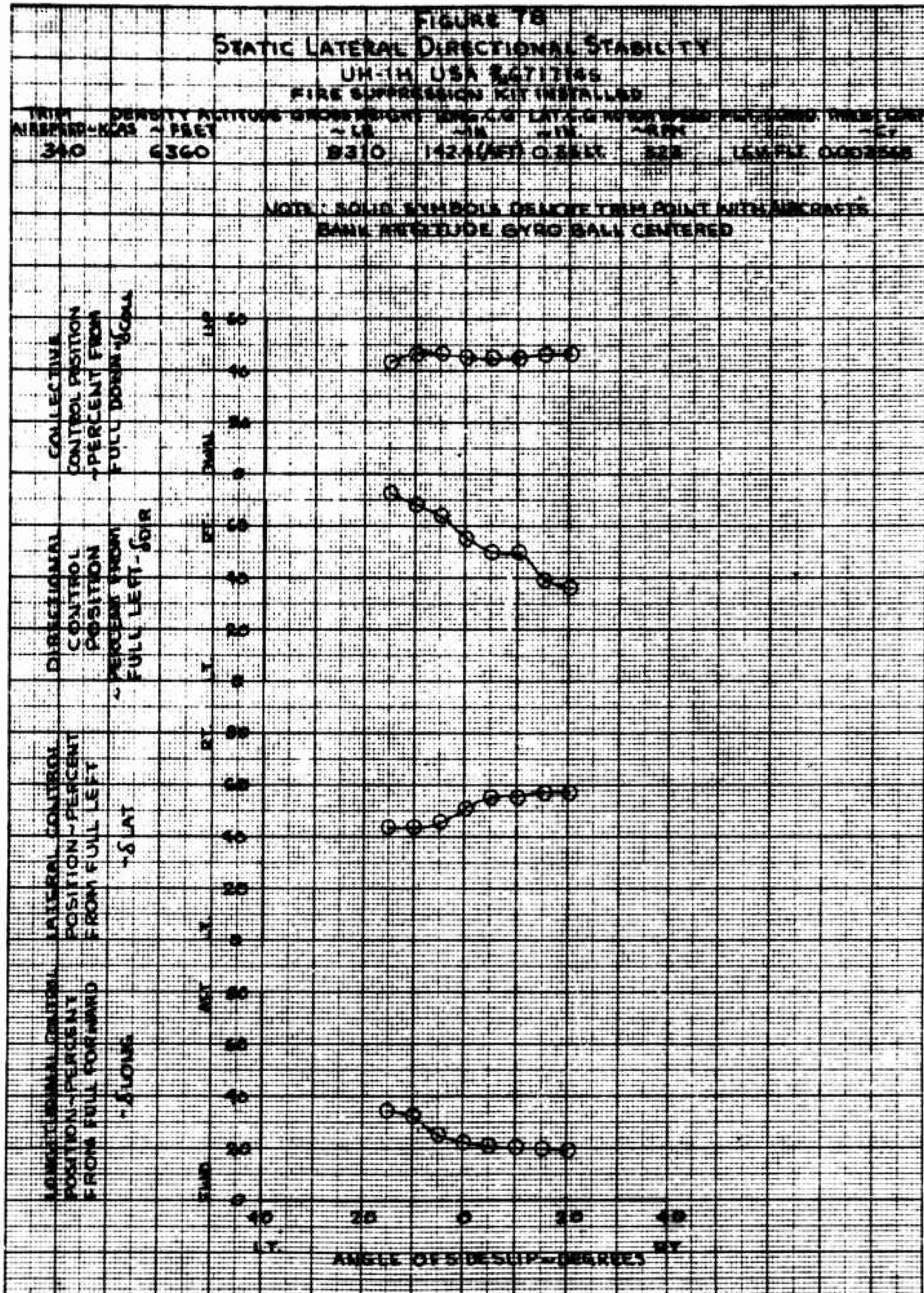


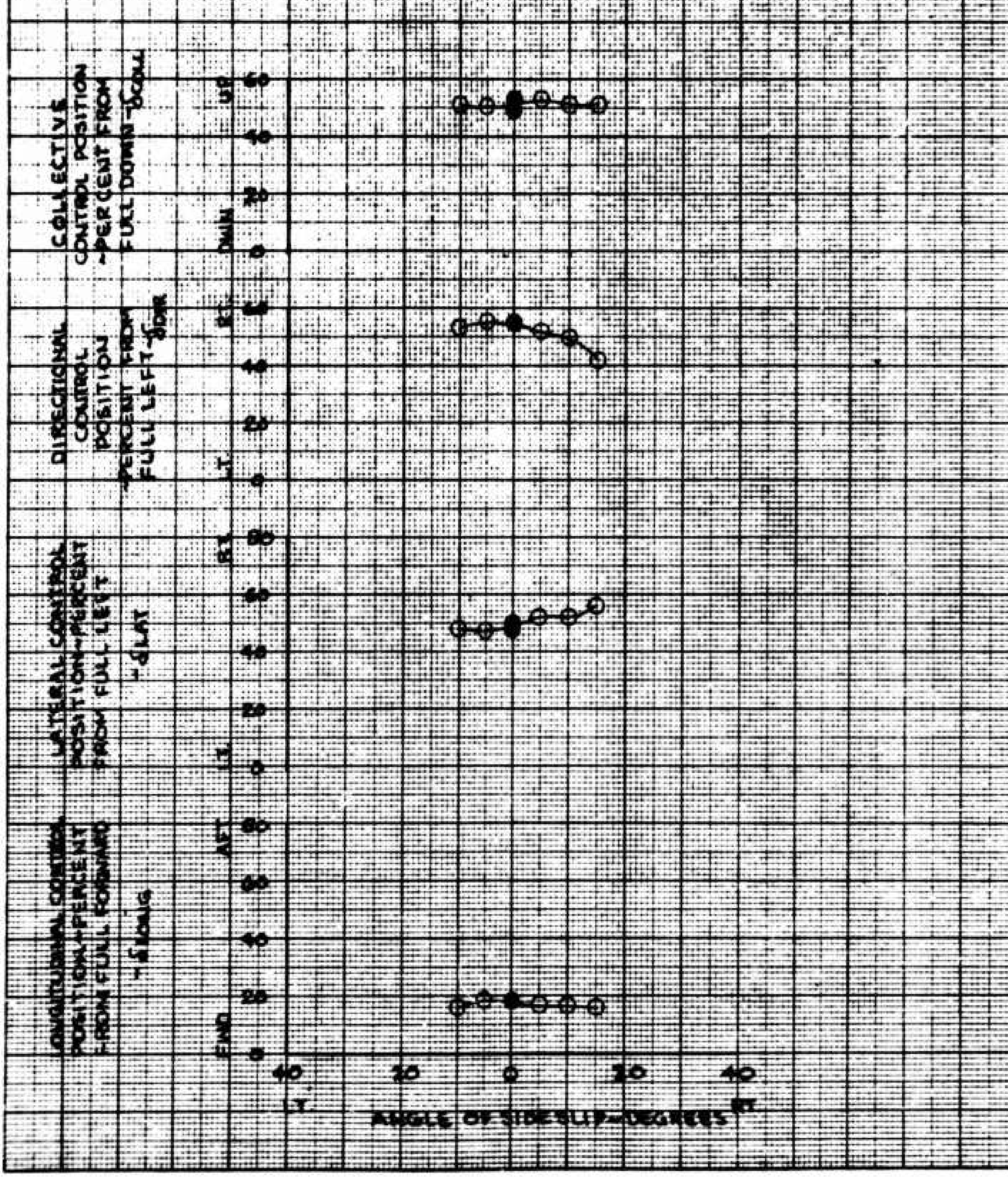

FIGURE 78
STATIC LATERAL DIRECTIONAL STABILITY
UH-1H USA S/N 67-17146
FIRE SUPPRESSION KIT INSTALLED

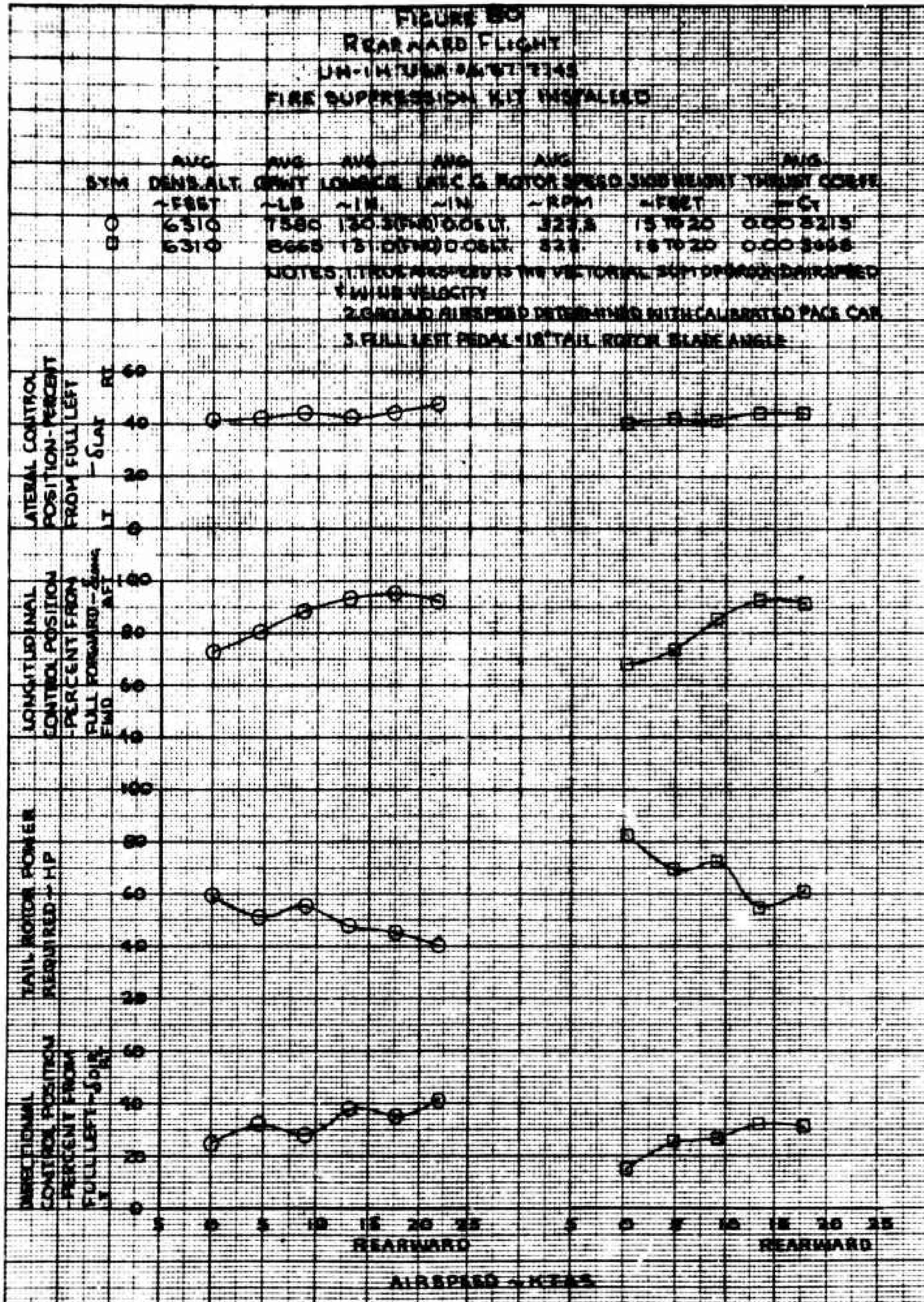
NOTE: SOLID SYMBOLS DENOTE TRIM POINT WITH AIRCRAFT'S BANK ATTITUDE GYRO BALL CENTERED

FIGURE 79
STATIC LATERAL DIRECTIONAL STABILITY
UH-1H USA S/N 67-17146
FIRE SUPPRESSION KIT INSTALLED

| TRIM CALIBRATED AIRSPEED~KCAS | DENSITY ALTITUDE ~FEET | GROSS WEIGHT ~LB | [illegible] | [illegible] | ROTOR SPEED ~RPM | [illegible] | THRUST COEFFICIENT ~$C_T$ |
|---|---|---|---|---|---|---|---|
| 87.0 | 6430 | 8198 | 147.0 | 0.9613 | 324 | LEVEL FLT | 0.003[illegible] |

NOTE: SOLID SYMBOLS DENOTE TRIM POINT WITH AIRCRAFTS BANK ATTITUDE GYRO BALL CENTERED




REARWARD FLIGHT
FIRE SUPPRESSION KIT INSTALLED
SYM
AVG. DENS. ALT. ~FEET
6310
6310
AVG. ROTOR SPEED ~RPM
15 TO 20
15 TO 20
3. FULL LEFT PEDAL
LATERAL CONTROL POSITION-PERCENT FROM FULL LEFT
LONGITUDINAL CONTROL POSITION -PERCENT FROM FULL FORWARD
TAIL ROTOR POWER REQUIRED - HP
DIRECTIONAL CONTROL POSITION -PERCENT FROM FULL LEFT
REARWARD
REARWARD
AIRSPEED ~ KTAS

## APPENDIX G. SYMBOLS AND ABBREVIATIONS

| Abbreviation | Definition | Unit |
|---|---|---|
| ALT | Altitude | foot |
| CG, cg | Center of gravity | -- |
| DWN | Down | -- |
| fig., figs. | Figure, figures | -- |
| FLT | Flight | -- |
| FS | Fuselage station | inch |
| fwd | Forward | -- |
| GRWT, grwt | Gross weight | pound |
| HQRS | Handling Qualities Rating Scale | -- |
| IGE | In ground effect | -- |
| KCAS | Knots calibrated airspeed | knot |
| KTAS | Knots true airspeed | knot |
| LT, lt | Left | -- |
| LONG. | Longitudinal | -- |
| NACA | National Advisory Committee for Aeronautics | -- |
| NO., no. | Number | -- |
| OGE | Out of ground effect | -- |
| PSI, psi | Pound(s) per square inch | $lb/in.^2$ |
| RPM, rpm | Revolution(s) per minute | rpm |
| RT, rt | Right | -- |
| SHP, shp | Shaft horsepower | -- |

| Abbreviation | Definition | Unit |
|---|---|---|
| SL | Sea level | – |
| S/N | Serial number | -- |
| STD, std | Standard | – |
| TRQ | Engine output torque | ft-lb |
| WT | Weight | pound |

| Symbol | Definition | Unit |
|---|---|---|
| A | Rotor disc area | $ft^2$ |
| a | Speed of sound | ft/sec |
| $C_P$ | Power coefficient | -- |
| $C_T$ | Thrust coefficient | – |
| f | Equivalent flat plate area | $ft^2$ |
| h | Skid height | foot |
| $H_D$ | Density altitude | foot |
| $H_P$ | Pressure altitude | foot |
| $l_t$ | Distance from center line of main rotor shaft to center line of a 90-degree gearbox output shaft | foot |
| M | Mach number | – |
| $N_E$ | Engine speed | rpm |
| $N_R$ | Main rotor speed | rpm |
| $N_{TR}$ | Tail rotor speed | rpm |
| P | Engine output torque pressure | in. of Hg |
| R | Rotor radius | foot |
| T | Temperature | °F, °C |

| Symbol | Definition | Unit |
|---|---|---|
| $V_{cal}$ | Calibrated airspeed | knot |
| $V_H$ | Maximum airspeed for level flight | knot |
| $V_L$ | Limit airspeed | knot |
| $V_T$ | True airspeed | knot |
| °C | Degree(s) Centigrade | degree |
| °F | Degree(s) Fahrenheit | degree |
| $\Delta$ | Difference | – |
| $\delta_{COLL}$ | Collective control position | inch |
| $\delta_{DIR}$ | Directional control position | inch |
| $\delta_{LAT}$ | Lateral cyclic control position | inch |
| $\delta_{LONG.}$ | Longitudinal cyclic control position | inch |
| $\mu$ | Main rotor tip speed ratio | – |
| $\rho$ | Air density | $slug/ft^3$ |
| $\sigma$ | Density ratio | – |
| $\Omega$ | Rotor rotational frequency | rad/sec |

| Subscript | Definition |
|---|---|
| a | Ambient |
| ENG | Engine |
| std, s | Standard |
| t | Test |
| TR | Tail rotor |
| MR | Main rotor |
| tip | Main rotor tip |

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13. ABSTRACT

Engineering flight tests were conducted on the UH-1H helicopter to evaluate the performance and handling qualities during hover, translational flight, and forward flight. Tests were conducted in California at Edwards Air Force Base and at test sites near Bishop during the period 13 July to 9 August 1971. The UH-1H helicopter is being purchased by the US Air Force to perform search and rescue missions. Selected performance parameters and handling qualities were quantitatively and qualitatively evaluated. For the conditions tested, the UH-1H does not comply with paragraphs 3.2.1, 3.3.2, and 3.3.6 of the military specification, MIL-H-8501A. There were three deficiencies, the correction of which appears essential for adequate mission accomplishment: (1) insufficient longitudinal control within the approved gross-weight/center-of-gravity envelope, (2) insufficient directional control, and (3) directional instability between 10 and 18 knots at relative azimuths between 210 and 320 degrees.

DD FORM 1 NOV 65 1473 REPLACES DD FORM 1473, 1 JAN 64, WHICH IS OBSOLETE FOR ARMY USE.

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| | ROLE | WT | ROLE | WT | ROLE | WT |
| UH-1H helicopter<br>Evaluate performance and handling qualities<br>Perform search and rescue missions<br>Performance parameters<br>Handling qualities<br>Qualitatively and quantitatively evaluated<br>Does not comply<br>Three deficiencies | | | | | | |

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