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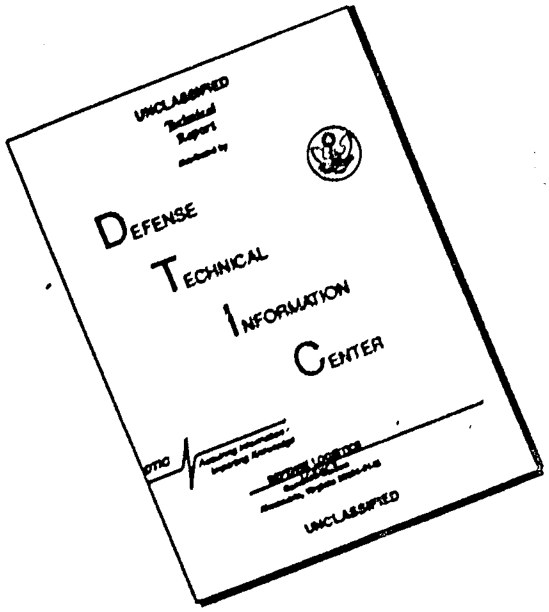

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ENGINEERING FLIGHT TEST

OF THE

LIGHT OBSERVATION HELICOPTER (LOH) OH-6A

ARMED (XM-27E1) AND UNARMED

PERFORMANCE AND STABILITY AND CONTROL

WILLIAM E. CROUCH, JR
LT COLONEL, ARTY
PROJECT OFFICER

EDWARD P. PREISENDORFER
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PROJECT ENGINEER

DECEMBER 1967

US ARMY AVIATION TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA

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ABSTRACT

A limited evaluation of the OH-6A aircraft was conducted to provide data for the Light Observation Helicopter (LOH) Competition. The primary effort was devoted to an investigation of the performance and stability and control characteristics of the OH-6A in its two mission configurations (Configuration I - unarmed, gross weight of 2290 pounds and Configuration II - armed with the XM-27E1 subsystem, gross weight of 2500 pounds). The performance data are quantitative and encompass hovering flight, climbing flight, level flight, and engine characteristics. All contractual performance guarantees were met, and the test engine surpassed the performance of the specification engine. The stability and control data are both quantitative and qualitative.

All of the handling qualities investigated met the requirements of MIL-H-8501A. Those paragraphs not investigated either appeared to meet the requirements or were not objectionable; and in the pilot's opinion, all handling qualities were generally good.

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INTRODUCTION

BACKGROUND

1. The engineering test program discussed in this report is a part of the second light observation helicopter evaluation conducted by the U.S. Army. The first evaluation was conducted in 1964, and resulted in the procurement and introduction into service of the OH-6A helicopter. The second evaluation consisted of a combined aircraft engineering and limited operational suitability program to compare the relative merits of the submitted aircraft. The ultimate goal of this effort is to provide data to assist in procuring the light observation helicopter most suited to Army requirements.

TEST OBJECTIVES

2. The limited test program covered in this report was conducted to make a rapid but comprehensive evaluation of the basic aircraft qualities with respect to accomplishment of the assigned mission. Specifically, the tests were accomplished to determine if the performance was commensurate with the General Specification requirements and if the handling qualities were adequate to meet both the General Specification and the intended mission requirements. In addition, limited maintainability information was obtained.

DESCRIPTION

3. The OH-6A is a turbine-powered single-rotor helicopter manufactured by the Aircraft Division of Hughes Tool Company. The OH-6A is powered by an Allison T63-A-5A gas turbine engine, derated to 225 normal rated horsepower at 104 percent RPM. Thrust and control are provided by a four-bladed, fully articulated main rotor plus an antitorque tail rotor. Three fixed airfoils are located on the tail to provide stability during high-speed forward flight. The flight controls consist of dual pedals, collective, and cyclic sticks. The controls are unboosted, but incorporate an electrically operated spring to trim out longitudinal and lateral cyclic stick forces. Adjustable friction controls are provided for both the collective and cyclic stick.

4. The OH-6A has a two-place, side-by-side cockpit configuration with seats for a pilot and an observer. A cargo area is located directly behind the cockpit seats, is accessible through two rear cargo doors, and can accommodate two passengers on stowable seats or equivalent cargo. The landing gear is of the skid type with air-oil shock struts. The basic physical characteristics and photographs of the OH-6A are shown in Appendix I. Additional descriptive information can be obtained from the Manufacturer's Detail Specification HTD-AD-369-Y-8011. The OH-6A currently in use by the U.S. Army is described in the Operator's Manual, TM55-1520-214-10.

SCOPE OF TEST

5. A total of 26 productive flight tests in 21 hours of flying time was performed to evaluate the performance and stability and control characteristics of the test aircraft. Performance tests were separated into three categories: (1) hovering, (2) level flight, and (3) climb. An airspeed calibration was also conducted. Hovering tests were conducted to derive C_P versus C_T plots. These data, in addition to the separately evaluated engine characteristics, were used to determine the degree of conformance with the hovering requirements in the General Specification. Level flight tests were conducted to find power and fuel requirements as a function of true airspeed. Climb tests were conducted at normal and military rated power to determine rate of climb and service ceiling. Conformance to range, endurance, and rate of climb criteria was determined from these tests.

6. Stability and control testing was done to insure flight safety and conformance with MIL-H-8501A. Static trim curves were obtained from the level flights. In addition, longitudinal collective-fixed stability and lateral-directional stability were evaluated at various trim points. Sideward and rearward flights were conducted to determine the control margins remaining within the specified speed range. Dynamic stability and controllability were evaluated by observing the aircraft response to simulated disturbances and control step inputs. Instrumentation was installed to record the various aerodynamic angles, rates, and accelerations.

7. The OH-6A was tested against the performance and stability and control requirements listed in the General Specification for the Light Observation Helicopter, dated 25 July 1967 (Reference 1, Appendix VII).

8. The performance requirements are listed in Paragraph 3.1.2.2 of the General Specification. The guarantees verified in this test program for Configuration I were as follows:

a. Power required, 110 KTAS, sea level - no greater than NRP (223 HP).

b. Maximum rate of climb at MRP (281 HP), sea level - at least 1500 fpm.

c. Hover ceiling at MRP, OGE, 95°F - at least 2000 feet.

d. Hovering ceiling at MRP, IGE, 4-foot skid height, 95°F - at least 5000 feet.

e. Range (takeoff fuel allowance 2 minutes at NRP, sea level and reserve fuel - 10 percent of initial fuel) - at least 260 nautical air miles.

f. Cruise speed for (e) above at NRP or less - .99 speed for best range (high side).

g. Endurance (at conditions for (e) above) - at least 3.0 hours.

The guarantees verified for the armed Configuration II were as follows:

a. Range (at conditions for (e) above) - at least 230 nautical air miles.

b. Hover ceiling at MRP, OGE, standard temperature - at least 6000 feet.

9. In addition, the manufacturer's estimates listed in HTC-AO-369-Y-8011 and requested in the General Specification were checked by the data obtained in the three categories of performance tests mentioned above.

10. The stability and control requirements are listed in Paragraph 3.3.2 of the General Specification. Testing was performed to check the aircraft compliance with MIL-H-8501A at all gross weights. Design speeds of 35 knots true airspeed (KTAS) sideward and 30 KTAS rearward are specified in Section 3.4.4, and tests were conducted to evaluate the flight characteristics at these speeds.

METHODS OF TEST

11. Performance flight tests were conducted at Edwards Air Force Base, Bakersfield, and Bishop, California, to provide a range of density altitudes. All tests were conducted in stabilized, nonturbulent air to insure that accurate repeatable performance data were obtained. Sensitive instruments were used and data were recorded manually and by a photo panel. Power available and fuel flow data, as specified in the T63-A-5A Engine Model Specification No. 580-F, were used to derive standard performance conditions. In addition, fuel flow was monitored in flight to check the test engine performance and to give an accurate accounting of the gross weight. Power required was derived from test engine calibrations plus tachometer and torquemeter indications.

12. Stability and control testing was accomplished at Bakersfield and Bishop, California. The majority of the testing was at a density altitude of approximately 6000 feet and at the most aft center-of-gravity (C.G.) location obtainable. Trim curves were obtained during level flight performance testing. Longitudinal collective-fixed stability, lateral-directional stability, dynamic stability, and controllability tests were flown in that order. The last three tests were flown in both mission configurations. Sideward and rearward flights were made in ground effect (IGE) at sea level at Bakersfield, California, in Configuration II with a forward C.G. location. Accurate airspeeds were obtained by use of a calibrated pace vehicle and an accurate windspeed determination. An oscillograph was used for all stability and control tests to record stick positions, aerodynamic angles, and aircraft motions. All test instrumentation installed in the aircraft is listed in Appendix II.

CHRONOLOGY

13. The chronology of the tests was as follows:

Test Directive Assignment Received	26 August 1967
Test Aircraft Received	9 October 1967
Tests Started	7 November 1967
Tests Completed	1 December 1967
Report Submitted	8 December 1967

RESULTS AND DISCUSSION

PERFORMANCE

General

14. Performance tests were conducted to confirm the contractor's guarantees and to insure compliance with the General Specification. Testing was performed in accordance with the standard procedures presented in Appendix III, and at the mission gross weights specified in Appendix IV. All weight and performance guarantees (Paragraphs 15, 19, and 30) were met.

Weight Empty

15. The guarantee empty weight was 1190 pounds and the actual weight adjusted to the guarantee conditions was determined to be 1190 pounds. The adjustment of the delivered weight to the guarantee condition is shown in Appendix IV.

Airspeed Calibration

16. The test boom and standard airspeed system were calibrated to determine the position error as a function of airspeed. The boom airspeed system position error is shown in Figure 1, Appendix V. The error varies linearly from + 3 knots at low speed (20-50 KCAS) to + 4 knots at high speed (70-120 KCAS). The characteristics and magnitude of error are comparable to those of other boom airspeed systems mounted on previously tested OH-6A aircraft.

17. The standard airspeed system position error is shown in Figure 2, Appendix V. The error is negligible at speeds above 70 KCAS and increases nonlinearly as calibrated airspeed decreases. The maximum error is + 6 knots at an indicated airspeed of 20 knots. This large error is undesirable, but is found in many other helicopter standard systems. Generally,

the airspeed system is satisfactory and has the desirable characteristic of being accurate in the cruise airspeed range.

Hovering

18. Hovering tests were conducted to evaluate the hovering performance of the OH-6A helicopter and to determine compliance with the contractual guarantees. Standard and hot day hovering performance at the two skid heights is shown in Figures 3 and 4, Appendix V. The test results are presented graphically in nondimensional form for skid heights of 4 feet IGE and 55 feet OGE in Figures 5 and 6, Appendix V.

19. The OH-6A met the three contractual guarantees by the margins shown in Table 1. The summary performance was derived from the nondimensional test results (Figures 5 and 6, Appendix V) and the plot of shaft horsepower (SHP) available as a function of pressure altitude and temperature (Figure 22, Appendix V). On a 95°F day, the available horsepower is limited by turbine outlet temperature. Power available thus decreases rapidly with altitude, resulting in a rapid decrease of maximum hovering gross weight as density altitude increases.

TABLE 1

OH-6A HOVER PERFORMANCE

Hover Ceiling	Guaranteed (ft)	Actual (ft)
OGE, Standard Temperature, Configuration II	6000	6530
IGE, 95°F, Configuration I	5000	5130
OGE, 95°F, Configuration I	2000	2970

20. On a standard day, the aircraft is torque-limited up to an altitude of 5000 feet. The decrease in rotor efficiency with altitude reduces the maximum hovering gross weight slightly from sea level to the critical altitude. This effect is also seen from the nonlinear increase of C_P as C_T increases. Above 5000 feet, the available power is limited by the maximum allowable turbine outlet temperature and the maximum hovering gross weight decreased rapidly. The decrease is less than that realized under hot day (95°F) conditions due to the beneficial ambient temperature gradient existing on a standard day.

21. The helicopter was easily stabilized with minimum pilot effort during IGE hover tests at wind conditions of 0-3 knots. At higher wind conditions (4-6 knots) the helicopter becomes sensitive to small gust spreads which induces a yawing or fish-tailing effect. Hover OGE is more easily stabilized than hover IGE. Pilot Rating Scale (PRS) A-3.

22. The cyclic force trim was effective in neutralizing stick forces in a hover. Control power appeared to be excellent on all axes. Minimum control (cyclic and collective) movements were required to maintain a stabilized hover. RPM control (beep) was easily controlled or changed with the left thumb while maintaining positive collective control. Small pedal changes while controlling heading cause very little torque changes up to maximum operating limits. (PRS A-2)

Climbs

23. Climb tests were conducted to determine the performance of the OH-6A helicopter during climbing flights at all altitudes, and to insure compliance with the contractual guarantees. Climbs were conducted at maximum continuous power, takeoff power, and various gross weights to determine the variation of rate of climb with power and gross weight.

24. Test results are shown in Figures 7 through 9, Appendix V. The actual rate of climb was corrected to the rate of climb that would have been achieved on a standard day with the power available shown in Figures 21 and 22, Appendix V. The actual climb schedules, distance traveled, and specification fuel flow are also shown. Variations of rate of climb with power and gross weight, and the resulting rate of climb correction factors are shown in Figure 7, Appendix V.

25. The OH-6A met its contractual guarantee of a sea level rate of climb of 1500 feet per minute (fpm) in Configuration I at takeoff power. The actual rate of climb at sea level was 1860 fpm; this increased to 1950 fpm at 4900 feet where the turbine-outlet-temperature (TOT) limit was reached. The rate of climb decreased rapidly above this density altitude due to the decrease in power available.

26. The sea level rate of climb in Configuration I at maximum continuous power was 1250 fpm. This rate of climb was roughly constant up to the critical altitude of 7600 feet where the TOT limit was reached. The rate of climb then decreased rapidly as the service ceiling of 16,600 feet was reached. At high altitudes, the rate of climb is temperature sensitive due to the small differences between power available and power required for level flight. This sensitivity is also apparent in the high values of K_W at large values of C_T (high density altitudes and high gross weights).

27. No significant power management problems were encountered during the climb tests. Climbs were entered from level flight, and engine response and acceleration characteristics were satisfactory. Considerable attention must be given to longitudinal control in climbs to maintain a desired airspeed because of the oversensitivity or strong control power along the longitudinal axis. (PRS A-3)

28. The test helicopter had a slight vertical vibration throughout all tests. During climb to service ceiling tests, the vibration intensity increased noticeably at a density altitude of approximately 13,500 feet and remained constant to service ceiling. Control response remains excellent about all axes to service ceiling. (PRS A-3)

Level Flight

29. Tests were conducted in level flight to determine the range, endurance, and power required by the OH-6A helicopter. Tests were also conducted with the XM-27E1 weapon system installed to determine its effect on the helicopter's performance. Data from these tests were used to check the contractual guarantees and to compute climb schedules. The test results are presented graphically in Figures 12 through 20, and are summarized nondimensionally in Figure 11, Appendix V. Nautical air miles per pound of

fuel (NAMPP) and airspeed are also plotted as a function of gross weight (see Figure 10, Appendix V).

30. Table 2 shows the contractual guarantees and the actual test results. All guarantees were calculated with the aircraft loaded to achieve a forward C.G. location (Station 97). This forward C.G. location resulted in generally lower performance than was estimated by the manufacturer; however, the helicopter did meet all of the guarantees. The deterioration in performance is shown in Figure 12, Appendix V. An additional 5 to 10 horsepower is required above 60 KTAS as C.G. location is moved from a mid to a forward location. This is due to the additional downward pitch attitude caused by the shift in C.G. and results in an approximately 10-percent decrease in range and endurance from the mid C.G. flight value. At the Configuration I gross weight of 2290 pounds, 221 SHP was required to fly at sea level at the guaranteed 120 KCAS. At takeoff power (281 SHP), a speed of 130 KCAS was achieved.

TABLE 2

OH-6A CONTRACT GUARANTEES

Configuration I	Guaranteed	Actual
V_{NE} (FAA certified S.L.)- KCAS	120	124
Power required at 110 KTAS S.L.- SHP	210	190
Range (including allowances) S.L.- nautical miles	260	268
Cruise speed (speed for .99 best range) - KTAS	105	116
Endurance (including allowances) S.L.- hours	3.00	3.06
Configuration II		
Range (including allowances) S.L.- nautical miles	230	250

31. Range and endurance calculations were based on a Configuration I fuel load of 406 pounds (Appendix IV) minus 10-percent reserve (40.6 pounds) and 2 minutes of fuel at sea level and maximum continuous power (5.6 pounds). The resulting useable fuel (360 pounds) gives a range of 268 nautical miles at a cruise speed of 109 KTAS. The speed for .99 best range (high side) is 116 KTAS. Power required for this speed is 209 SHP, which is less than maximum continuous power (223 SHP). Maximum endurance is achieved at a loiter speed of 62 KTAS at sea level. The endurance is 3.06 hours for the same useable fuel load of 360 pounds. The maximum range and endurance are 274 nautical miles and 3.17 hours respectively, with a full fuel load of 418 pounds and a useable fuel load (under the same conditions as above) of 371 pounds.

32. The fuel load of 383 pounds in the armed configuration (2500 pounds) results in 341 pounds of useable fuel under the same conditions as for Configuration I. This amount of fuel gives a range of 250 nautical miles at a cruise speed of 102 KTAS. For a full fuel load under the same conditions, a range of 272 nautical miles can be achieved. Figures 19 and 20, Appendix V, show the level flight performance of the OH-6A in Configuration II and include a comparison plot of the Configuration I performance (dotted line) at the same value of C_T. At speeds above 70 KTAS, the weapon drag results in a 3- to 8-percent increase in shaft horsepower required and approximately a 3- to 8-percent decrease in range and endurance. The limited data suggest that the effect of the weapons drag is greater at higher values of C_T due to its effect of decreasing the width of the characteristic speed-power "bucket" curve. At all density altitudes tested (sea level to 14,000 feet in Configuration I, and sea level to 5000 feet in Configuration II), there is sufficient power available to reach V_{NE}.

33. No unusual flying qualities characteristics were noted during the level flight performance tests. As V_{max} airspeeds were approached, a distinct increase in airframe vibrations was encountered. This was first felt in the cyclic stick, and as intensity further increased, feedback could be felt in the collective stick. No pitch-up or rolling tendencies were noted as the helicopter was approaching blade stall conditions. (PRS A-2)

34. Power management during level flight performance did not present any unusual problems. The droop cam compensator provided near instantaneous reactions to changes in power requirements. Less than 1 percent rotor RPM variation at 483 rotor RPM was encountered in stabilized level flight throughout the airspeed range tested.

Engine Characteristics and Inlet Data

35. Engine data are shown in Figures 21 through 27, Appendix V. Specification engine, calibrated engine, and actual engine data taken in flight are presented. The power available curves (Figures 21 and 22, Appendix V) are based on a specification engine (Reference 4) with the actual inlet as defined from flight test data. At takeoff power, the maximum allowable turbine outlet temperature is 749°C. The installed engine is torque-limited with a maximum allowable toruqe of 169 inches of mercury (in. Hg) which limits the shaft horsepower available to 278 SHP at 103 percent RPM (483 RPM), and to 281 SHP at the newly certified 104 percent RPM (489 RPM). The additional horsepower, caused by increasing rotor speed to 104 percent, increases the takeoff power sea level rate of climb by 35 fpm, but does not increase the hover ceiling of either Configuration I or II, since the engine is TOT limited in that range.

36. At maximum continuous power, the maximum allowable turbine outlet temperature is 693°C and the installed engine torque limit is 127 in. Hg. The shaft horsepower available is limited to 221 SHP at 103 percent RPM and to 223 SHP at 104 percent RPM. In the region where the engine is TOT limited, the actual inlet characteristics determine the available horsepower as a function of altitude and temperature. The actual inlet (Figure 23, Appendix V) showed no ram pressure rise, and gave a constant pressure ratio P_2/P_a (inlet pressure/ambient pressure) equal to .995 for all speeds. There was a temperature rise T_2-T_a (inlet temperature-ambient temperature) of 2°C at all speeds. These two detrimental effects (reducing pressure and increasing temperature) reduced the available power by about 5-1/2 SHP at any given altitude and ambient temperature. This resulted in a corresponding decrease of about 50 pounds of weight which could be carried while hovering under any TOT limited condition.

37. The calibrated engine was better than the specification engine with respect to the TOT being lower under any given conditions. The actual engine in flight had a performance midway between the specification and calibrated engines (Figures 26 and 27, Appendix V). Fuel flow and gas producer speeds as a function of shaft horsepower are shown in Figures 24 and 25, Appendix V. Fuel flow was not measured in flight, but the horsepower versus gas producer speed data taken in flight agreed with both the specification and calibrated engines.

STABILITY AND CONTROL

General

38. Stability and control testing was done to evaluate the handling qualities of the OH-6A and to check compliance with MIL-H-8501A. Testing was performed in accordance with the standard procedures presented in Appendix II and at the mission gross weights specified in Appendix IV. Due to the limited time available for flight testing, all the requirements of MIL-H-8501A could not be tested. None of the quantitative tests made showed any deviation from these requirements. The remaining requirements were investigated qualitatively or not at all, and are discussed in the Compliance with Flying Qualities Specification Section (Paragraphs 64 and 65). An overall Pilot Rating Scale (PRS) of A-2 is assigned to the handling qualities of the OH-6A helicopter.

Static Trim Stability

39. Static trim stability tests were conducted to determine control position trim curves for the OH-6A helicopter. The trim stability data were obtained for Configuration I at altitudes from sea level to 14,000 feet and at a rotor speed of 483 RPM. The C.G. location varied from most forward to most aft. The speed range was from 30 KCAS to the maximum allowable. A limited evaluation was accomplished for Configuration II. Test data are shown in Figures 29 through 31, Appendix V. The cyclic control pattern is shown in Figure 28, Appendix V.

40. The longitudinal trim stability was positive (forward stick position for a higher airspeed) for Configuration I with no control reversals from 25 to 125 KCAS and there were no significant discontinuities. More than 10 percent control margin was available at the maximum airspeed limit. Increased altitude required more forward stick, and the maximum change recorded was 1.5 inches at 68 KCAS for an altitude variation from 1000 to 13,800 feet. The increased forward stick requirements became greater with higher airspeed. Moving longitudinal C.G. from aft to forward introduced a requirement for approximately 1.5 inches of forward stick at 60 KCAS. This differential tended to decrease slightly at higher airspeeds.

41. The collective stick, lateral stick, and pedal requirements following the power required curves are shown in Figures 29 through 31, Appendix V. The high power settings required the most left pedal and left stick, with the maximum variation being approximately 1 inch. (PRS A-2)

42. The Configuration II characteristics were similar with the exception of the right lateral stick shift required to compensate for the lateral C.G. change caused by the weapon installation. (PRS A-2)

Static Longitudinal Collective-Fixed Stability

43. Static longitudinal collective-fixed stability tests were conducted for Configuration I (unarmed and 2290 pounds) at a density altitude of 6000 feet, a rotor speed of 483 RPM, and the most aft C.G. location obtainable (Station 101.3). The trim airspeeds tested were 35 KCAS, .8 V_{max} (81 KCAS), and V_{max} (102 KCAS). Test data are shown in Figures 32 through 34, Appendix V.

44. The static longitudinal collective-fixed stability (stick fixed) was positive (forward stick required to increase airspeed) for the trim speeds investigated. The stability characteristic was essentially linear with a slight tendency for reduced stability below the trim speed and increased stability above the trim speed. The stability level increased slightly as airspeed was increased from .8 V_{max} to V_{max}. (PRS A-2)

45. The control force (stick free) stability was similar to the stick fixed stability (increased forward pressure required to increase airspeed). The level of stability was not determined quantitatively. No unusual lateral or directional control requirements were introduced by a change from the trim speed. Generally, decreasing airspeed required left pedal and left lateral cyclic control, with the opposite being required for an increase in airspeed. (PRS A-2)

Static Lateral-Directional Stability

46. The static lateral-directional stability tests were conducted in Configuration I and II at the most aft obtainable C.G. location, a rotor speed of 483 RPM, and a density altitude of 6000 feet. The trim speeds investigated were 35 KCAS, .8 V_{max}, and V_{max}. The sideslip envelope tested was 15 degrees at maximum airspeed and varied linearly to 45 degrees at 35 KCAS.

47. The static directional stability for Configuration I was positive for all conditions tested and became stronger with higher airspeeds. At speeds above .8 V_{max}, the stability was essentially linear and there was sufficient directional control power to exceed the maximum specified sideslip angle. At the 35 KCAS condition, the stability was nonlinear, particularly at high right sideslip angles. Effective dihedral was positive and became stronger with increased airspeed. The characteristic was essentially linear for angles 20 degrees from trim. The static directional stability for Configuration II shows no significant difference from that of Configuration I. (PRS A-2)

Sideward and Rearward Flight

48. Sideward and rearward flights were conducted to determine the control positions required to hover under cross-wind and tail-wind conditions in Configuration II. Due to a lack of acceptable weather in the short time period allotted to the test flight phase, test results were not reduced in the OH-6A, Serial Number (S/N) 65-12927, but test data are presented for the OH-6A, S/N 65-12967, in Figures 35 and 36, Appendix V (Reference 7).

49. In sideward flight, the forward C.G. location was chosen, since it is the most critical in terms of control margins. No control limits were reached from 30 KTAS left to 30 KTAS right, and indications are that they would not be reached up to 35 KTAS in either direction. There were no pedal or lateral control reversals throughout the speed range. The most critical control was the longitudinal stick in left sideward flight above 10 KTAS. A reversal was present at 18 KTAS where 12 percent of aft longitudinal control remained. At no time was there ever less than 10 percent aft longitudinal control remaining. However, it has been noted (Reference 7) that this control is very sensitive to rigging changes. Also, flying in the unarmed configuration with a mid lateral C.G. location may result in less than 10 percent aft longitudinal control remaining. The contractor has a deviation (Reference 2) which requires only 20 KTAS in sideward and rearward flight. (PRS A-3)

50. Rearward flight at speeds above 25 KTAS required approximately the same degree of aft longitudinal control as left sideward flight. At no speed up to 35 KTAS rearward, however, was there less than 10 percent aft longitudinal control remaining. From 24 KTAS to 35 KTAS rearward, approximately 12 percent of aft longitudinal control was available. However, this is subject to the same rigging conditions as were sideward control position margins. There was a lateral and pedal control reversal at 27 KTAS rearward, but the magnitude of the reversal was small. There was no longitudinal control reversal, but the longitudinal control position remained constant from 25 KTAS. Speed was increased in this range by increasing collective. (PRS A-2)

Dynamic Stability

51. Dynamic stability tests were conducted for Configuration I and II at a rotor speed of 483 RPM, a density altitude of 5000 feet, and the most aft C.G. location obtainable. The speeds tested were 35 KCAS, .8 V_{max}, and V_{max}. The overall dynamic stability was satisfactory. Representative time histories for longitudinal, lateral, and directional pulses are shown in Figures 37, 38, and 39, Appendix V.

52. During hovering flight, the aircraft was not as stable as in forward flight, but still had acceptable dynamic stability characteristics. Following an aft longitudinal disturbance, the rate response was a heavily

damped oscillation with a period of 7 seconds. No small residual oscillations were present. The pitch attitude returned to the trim position and continued downward to reach its maximum nosedown value after about 5 seconds. Pitch attitude was damped back to normal after approximately one cycle.

53. A right lateral 1 inch pulse applied while hovering caused an oscillation of the yaw attitude and rate as well as the roll attitude and rate. These oscillations were well damped after about 6 seconds. A pedal input in hovering flight caused the aircraft to stabilize at a new heading with no residual oscillation. The roll produced by a pedal pulse was was complimentary and highly damped.

54. In forward flight, the longitudinal dynamic stability was very good at all speeds tested. Following an aft pulse of approximately 1/2 inch the pitch attitude returned to its trim value in 5 seconds after the input. The pitch attitude did not decrease below its original trim value. The pitch rate also returned to zero in 5 seconds after the pulse was applied but it experienced from 1 to 1-1/2 cycles before doing so.

55. The lateral dynamic stability was also good. Roll attitude oscillations were damped out in 1/2 cycle. The right lateral input caused a yaw to the right as well as a roll to the right, but the yaw oscillation was delayed for about 2 seconds after the input.

56. Directional dynamic stability at forward speeds was satisfactory. Right pedal inputs of 1/2 inch caused a right roll at all speeds. Directional dynamic stability showed a definite improvement with speed. Although lateral stick and pedal inputs showed aerodynamic coupling effects, the pitch attitude was not affected. An overall PRS of A-1 is assigned to the dynamic stability characteristics of the OH-6A helicopter.

Controllability

57. Controllability tests were conducted to determine the rates and displacements that result from various size control inputs. Test results are

shown in Figures 41 through 46, Appendix V, and summarized in Figure 40, Appendix V. Testing was generally done at 5000 feet density altitude and at a mid C.G. location in Configuration I.

58. The longitudinal control response (maximum initial angular velocity per inch of control travel) was satisfactory. In level flight (35 to 102 KCAS), the time to reach the maximum pitch rate following an aft longitudinal control input was 0.6 second. The pitch response in level flight was approximately 12 degrees per second for a 1 inch aft control input, and the plot of pitch response versus control input was nearly linear (Figure 41, Appendix V). In a hover, it took 1.1 seconds to reach the maximum pitch rate, which was greater (17 degrees per second for a 1 inch aft control input) than the maximum pitch rate for level flight. The magnitude of the angular pitch displacement 1 second after a 1-inch aft control pulse was approximately 7 degrees for both level flight and hover. This pitch attitude displacement varied approximately linearly with control displacement (Figure 41, Appendix V).

59. The longitudinal control response and angular pitch displacement for a forward C.G. in Configuration I and for a rearward C.G. in Configuration II (with weapon) can be seen in Figure 42, Appendix V. Neither the shift in C.G. location nor the addition of the weapon has any discernible effect on either the maximum pitch rate or 1-second pitch displacement for the two stabilized flight speeds chosen. At the higher speed (92 KCAS) in the armed configuration, however, it took longer to reach the maximum pitch rate (approximately 1 second) than it did for all the other level flight conditions. This could be due to the effective negative angle of attack of the weapon and support pod when the helicopter is in the pitch-down attitude of high speed forward flight. The resultant force forward of the rotor axis (the weapon is at Station 90) produces a downward pitching moment which opposes the pitch-up control moment during the initial portion of the pitch displacement. This delays the time to reach maximum rate, but does not appreciably decrease the maximum rate since the weapon moment changes sign as the helicopter pitches up.

60. Lateral control response was high but not objectionable at all forward speeds and in hovering flight. The average maximum roll rate was 19 degrees per second per inch of right control travel for high speed level flight and hovering flight, and was slightly lower for level flights at 35 KCAS (Figures 40 and 43, Appendix V). The time to reach maximum roll rate was approximately 0.5 second for all conditions. The roll attitude 1 second

after right control displacement varied linearly with the control displacement (Figure 43, Appendix V). It was approximately 12 degrees per inch of control travel in high speed level flight and hovering flight, and slightly lower in low speed flight. No significant changes were found in lateral control response or in the roll attitude after 1 second for either a forward shift of C.G. or for an increase of gross weight and addition of the XM-27E1 subsystem (Configuration II). This is illustrated in Figures 40 and 44, Appendix V.

61. Directional control response was much greater in hovering flight than in level flight, being 75 degrees per second per inch in a hover as opposed to 30, 21, and 18 degrees per second per inch in level flight at 35, 81, and 102 KCAS respectively. This maximum yaw rate in hovering flight took a relatively long time to develop (Figure 45, Appendix V)but was not considered objectionable by the pilot. In high speed flight, the time to reach maximum yaw rate was 0.6 second. At 35 KCAS, it was about 0.9 second and the time increased to between 1.5 and 3 seconds for hovering flight. The time to reach maximum yaw rate in a hover appears to be dependent upon the size of control displacement. (PRS A-3)

62. The yaw attitude 1 second after a 1-inch pedal displacement was 28 degrees for hovering flight and decreased to 18 degrees for 35 KCAS level flight, 14 degrees for 81 KCAS level flight, and 10 degrees for 102 KCAS level flight (Figure 45, Appendix V). These yaw attitudes varied linearly with pedal displacement and were considered satisfactory. (PRS A-3)

63. A shift to a forward C.G. location in Configuration I had no effect on the directional control response. In the armed configuration, the time to reach maximum yaw rate remained at 0.6 second in high speed level flight, but the control response to a right pedal input was reduced to 16 degrees per second per inch at 74 and 92 KCAS in level flight (Figures 40 and 46, Appendix V). This is due to the increased weight and the drag of the weapon. The yaw attitude 1 second after a right pedal input remained unchanged for the forward C.G. case but was reduced to 10 degrees for level flight at the above speeds in Configuration II. (PRS A-3)

Compliance with Flying Qualities Specification

64. The general flying qualities requirements are specified in Reference 12 and contractor requested deviations are presented in Reference 2. The limited scope of test did not include testing in all necessary configurations, flight regimes, and extremes of the flight envelope. The test instrumentation did not include vibration or control force measuring equipment. The vibration was evaluated solely by qualitative pilot comments while control force characteristics were determined by pilot comment.

65. The quantitative data and qualitative pilot comments indicate that the flying qualities are generally within the specified requirements and the requested deviations. Significantly more testing would be required to accomplish a detailed compliance determination.

MISCELLANEOUS

Cockpit Evaluation

66. The pilot enters the cockpit through the right front cockpit door. An assist handle is located on the topside of the door frame inside the door. With the cyclic stick attached to a torque tube at the base and front of the pilot's seat, some difficulty is encountered getting the left foot and leg to the inside position. However, after several times of entering and exiting, a pilot should not have any difficulty in rapid entrance or axit.

67. The pilot and observer seats are stretched mesh webbing on a frame for both seat and back-rest. Seats are comfortable but not adjustable and should not create any fatigue problem for duration of a full fuel load flying time (approximately 3 hours).

68. All switches, circuit breakers, and radio controls are easily reached by the pilot with the exception of the map light and switch located above the first aid kit on the center structure, left of the pilot's seat. Relocation of the map light and switch is desirable for improved Army use.

69. Cyclic, collective, and pedals are located so that they can be comfortably manipulated by the pilot sitting in an upright position. The directional pedals have three positions which will easily accommodate the average size pilot. Cyclic travel is restricted in the right rear position by pilot's right leg. The restriction was approximately 1 inch of right lateral travel to the project pilot but will vary with pilot size. The above evaluation was made in a static position on the ground and rotor not turning. No restrictions were encountered during actual flight testing and it is felt that no restrictions would be encountered in normal flight regimes. Correction is desirable for improved Army use.

70. The instrument panel layout was evaluated in the test helicopter and on Hughes Tool Company proposed arrangement (Drawing No. 369ASK427). The

general layout is satisfactory with the following exceptions:

a. Panel Arrangement:

(1) The helicopter was designed for a VFR mission; therefore, the airspeed indicator and N_2 rotor tachometer should be more directly in front of the pilot in the place of the altimeter and clock.

(2) The clock should be moved down to the fuel quantity location, the fuel quantity to the N_2 rotor tachometer location, and the altimeter to the present airspeed location.

(3) By placing the airspeed and N_2 rotor tachometer at the right edge of the panel reduces the pilot's scan of his primary VFR instruments and enhances his view outside the cockpit for the VFR mission. Although this is in conflict with the standard "T" panel layout, rearrangement is desirable for improved Army use.

b. Yellow/amber caution lights are not discernible when lit in bright sunlight. Correction is desirable for improved Army use.

71. The first aid kit and fire extinguisher are easily accessible to the pilot. Cockpit doors are jettisonable with an emergency cable handle only after doors have been opened with normal handle. The pilot's door handle is behind his right shoulder and difficult to reach with either hand. The emergency door release should be a single, easily accessible operation. Correction is required for satisfactory Army use. The visibility from either cockpit seat is generally good. The instrument panel creates a blind spot in the forward center cockpit area from the horizon down to the pedal locations but is not objectionable.

Helicopter Maintainability

72. The aircraft availability was excellent during this program. The aircraft was available for testing every day. The contractor maintenance personnel expended approximately 3 man-hours per day on the aircraft. This level of effort is appropriate for a test aircraft.

73. Maintenance problem areas encountered during the test program were as follows:

a. A small crack developed on the right side trailing edge of fairing above the tail pipe (aircraft hours - 139:00).

b. The external power plug bracket was broken while removing the APU cord (aircraft hours - 147:00).

c. Two main rotor dampers were replaced because their operation was not smooth and exceeded torque limits (aircraft hours-147:00 and 154:00).

d. The emergency release handle cable on the pilot's door frayed and broke off (aircraft hours - 151:35).

e. The sideslip ball race lost fluid causing the ball to be inoperative (aircraft hours - 154:00)

f. A small crack developed on the engine inlet air fairing (aircraft hours - 158:40).

g. The left and right position light brackets cracked (aircraft hours - 161:20).

h. The aircraft was delivered with a slight 1 per rev vertical vibration at all speeds. No attempt was made during the test program to correct this item as it was felt that it would not impair any qualitative or quantitative data.

Vibration and Noise Levels

74. Vibration test equipment was not available during the test period. Noise level requirements were deleted from the test program as a result of instructions issued from the U.S. Army Aviation Materiel Command. These tests had been previously conducted on a production aircraft by the U.S. Army Aeromedical Research Unit, Ft. Rucker, Alabama.

CONCLUSIONS

General

75. The aircraft and engine performance characteristics and flying qualities of the OH-6A helicopter met all the requirements within the scope of this test.

Deficiencies and Shortcomings Affecting Mission Accomplishment

76. Correction of the following shortcomings is desirable for improved helicopter operation and mission capabilities:

a. Map light and switch difficult to reach by pilot (Paragraph 68).

b. Restriction of right lateral cyclic travel in full aft position by pilot's leg (Paragraph 69).

c. Primary VFR instruments not properly located for mission (Paragraph 70).

d. Yellow caution lights not discernible when panel is in bright sunlight (Paragraph 70).

e. Emergency jettison of cockpit doors difficult (Paragraph 71).

f. Minor airframe and maintenance problems (Paragraph 73).

RECOMMENDATIONS

77. Correct those shortcomings stated in Paragraph 76, for which correction is desirable for improved helicopter operation and mission capabilities.

APPENDIX I

GENERAL AIRCRAFT INFORMATION

AIRCRAFT DIMENSIONS AND WEIGHTS

Overall Length	30	ft	3-3/4	in.
Fuselage Width	4	ft	6-3/4	in.
Skid Width	6	ft	9-1/4	in.
Overall Height	8	ft	4	in.
Empty Weight	1153			pounds
Configuration I (unarmed) Weight	2290			pounds
Configuration II (armed) Weight	2500			pounds
Overload Gross Weight	2700			pounds
C.G. Limits	97-104			in.

ENGINE AND TRANSMISSION LIMITS

Shaft Horsepower (TO power, 104 % RPM)	281	SHP
Shaft Horsepower (Maximum continuous power, 104% RPM)	223	SHP
Turbine Outlet Temperature (TO power)	1380°F(749°C)	
Turbine Outler Temperature (Maximum continuous power)	1280°F(693°C)	
Rotor RPM (104 %)	489	RPM

APPENDIX II

TEST INSTRUMENTATION

The test instrumentation used during this evaluation was supplied, installed, and maintained by the Logistics Division of the U.S. Army Aviation Test Activity. Calibration of the instrumentation was accomplished by the Instrumentation Branch of the Air Force Flight Test Center and the Logistics Division of the U.S. Army Aviation Test Activity.

A swivel-mounted pitot-static airspeed head was installed on a boom which extended approximately 5 feet in front of the nose of the helicopter. This airspeed head was used as a source for the sensitive altitude and airspeed systems. Vanes attached to the boom were used to measure angles of attack and sideslip.

Sensitive instrumentation was installed to measure the following parameters:

Pilot-Engineer Panel

Boom System Airspeed
Ship Airspeed
Boom Altitude
Outside Air Temperature
Rotor Speed
Torque Meter Pressure
Turbine Outlet Temperature
Turbine Compressor Speed
Fuel Counter

Photo Panel

Boom Airspeed
Boom Altitude
Outside Air Temperature
Rotor Speed
Torque Meter Pressure
Compressor INlet Temperature
Compressor Inlet Pressure (Altimeter)

Oscillograph

Pitch Angle
Pitch Rate
Yaw Angle
Yaw Rate
Roll Angle
Roll Rate
Collective Stick Position
Cyclic Stick Longitudinal Position
Cyclic Stick Lateral Position
Pedal Position
Angle of Attack
Angle of Sideslip
C.G. Normal Acceleration

APPENDIX III

METHODS OF TEST

AIRSPEED CALIBRATION

1. A level flight airspeed calibration was accomplished over a measured ground course to determine the position error of the test boom and standard airspeed system. Reciprocal headings were flown at the same indicated airspeed, and the average lapsed time was used to correct for wind velocity and direction. Pressure and temperature measurements were used to obtain the air density ratio. The basic calibration was conducted at the normal operation rotor speed and checked at the minimum rotor speed. Variations with ground proximity, gross weight, center of gravity location, and flight regime were not determined.

HOVERING PERFORMANCE

2. Hovering performance data were obtained using the free-flight method at skid heights of 4 feet (IGE) and 55 feet (OGE). A weighted cord of the desired length attached to the skid was used in conjunction with radio signals from the ground to maintain the aircraft at the desired height. Gross weight was varied by incrementally adding ballast while maintaining a mid C.G. location. Maximum and minimum rotor speeds were used at each height and weight. Testing was done at two density altitudes, Bishop (Approximately 4300 feet) and Bakersfield, California (approximately sea level), to further vary the range of power and thrust coefficients and to determine if any rotor deterioration with altitude was present. Each skid height was flown separately with rotor speed being varied to obtain two data points at each gross weight. Engine torque was recorded manually when the pilot indicated the aircraft was stable at the given skid height and rotor speed. Pressure altitude, free air temperature, and fuel counter readings were also recorded by hand, and the photo panel was used primarily in a backup capacity.

CLIMB PERFORMANCE

3. The best climb airspeed for various gross weights, rotor speeds, power, and altitudes were determined from the level flight performance data. Normal rated power climbs were accomplished from approximately sea level to the service ceiling. Military rated power climbs were terminated at 10,000 feet. During the climbs, power limits were established by the lowest allowable engine limit for the power setting.

4. Two climbs were made at each power setting. The pressure altitude, indicated airspeed, torquemeter reading, time, and fuel counter readings were recorded by hand every 500 feet. The photo panel was run at 1 frame every 2 seconds throughout the climb. The above climbs were all made at Mission I gross weights at a mid C.G. and were used to compute the power correction factor (K_P). Two additional climbs at normal rated power and gross weights of 2190 and 2450 pounds at mid C.G. were made to obtain a weight correction factor (K_W). These two factors were used to correct the climb data to standard day conditions.

5. The effects of airspeed and rotor speed variations were not investigated during the climb tests.

LEVEL FLIGHT

6. Level flight power required tests were conducted at Mission I gross weight for three different C.G. locations to determine the most adverse drag condition. The C.G. locations were chosen to cover the allowable range. Four additional flights were made at the least desirable C.G. location (forward) at different altitudes and weights necessary to vary the nondimensional coefficients. During each speed power flight, as fuel was consumed, the altitude was adjusted so that a constant ratio of weight to density (W/ρ) was maintained. Rotor speed was also held constant throughout, so that all the points on a specific curve were flown at a constant thrust coefficient. Pressure altitude, free air temperature, rotor

engine torque, turbine outlet temperature, and fuel used were manually recorded at each stabilized level flight speed. Speed was varied incrementally throughout the allowable speed range for each flight condition. In addition, the photo panel was used to provide engine characteristics and the oscillograph was used to establish the trim control positions. Tests were also conducted in the armed configuration to determine the performance and stability and control contributions of the weapons system.

ENGINE PERFORMANCE

7. The aircraft engine was calibrated on a test stand by the engine manufacturer prior to installation. Data were recorded to determine the installed performance for the ambient test conditions. Ambient and compressor inlet pressures and temperatures were recorded at various density altitudes for calibrated airspeeds ranging from 0 to 120 knots. The resultant data were used with the Engine Model Specification to determine the installed power available. The engine characteristics were determined from the relationships between gas producer speed, fuel flow, engine torque, and turbine outlet temperature.

STATIC TRIM STABILITY

8. Tests were conducted to determine the static longitudinal trim stability and flying qualities at a series of trim airspeeds during level flight. Stick positions were recorded on an oscillograph during the level flight testing when the aircraft was stabilized at a given speed.

STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY

9. The static longitudinal collective-fixed stability tests were conducted to quantitatively measure the static stability characteristics and flying qualities as airspeed was varied about a given

trim airspeed at a fixed collective setting. The aircraft was trimmed at the desired airspeed and the collective control was locked in position. Airspeed was then incrementally varied through the range specified in Reference 12. Cyclic and directional control was used as required to achieve the necessary speed changes. The sideslip angle was held at zero during the tests.

10. An oscillograph and a photo panel were used to record atmospheric conditions, control positions, aircraft positions, and motions. The instrumentation is presented in Appendix II.

STATIC LATERAL-DIRECTIONAL STABILITY

11. The static lateral-directional stability tests were conducted to determine the static directional stability and effective dihedral characteristics at various airspeeds and sideslip angles. The trim speeds and sideslip angles were as specified in References 5 and 12. The aircraft was trimmed at the desired airspeed and the collective stick was locked in position. The heading and airspeed were maintained constant while sideslip angles was varied both right and left. Data were recorded for each stabilized condition. The instrumentation used to record the data is listed in Appendix II.

SIDEWARD AND REARWARD FLIGHT

12. Sideward and rearward flight tests were conducted to determine the control positions required to hover under cross-wind and tail-wind conditions. These cross-winds and tail-winds were simulated by flying at certain sideward and rearward speeds. These speeds were measured by stabilizing IGE on a calibrated ground vehicle travelling at the desired speeds. When the aircraft was stabilized on the ground vehicle, the stick positions and aircraft attitudes were recorded on an oscillograph. The instrumentation used is listed in Appendix II.

DYNAMIC STABILITY

13. The dynamic stability characteristics were determined from the aircraft motions resulting from a dynamic disturbance. An external disturbance was simulated by introducing a control pulse into the individual axis being investigated. Control position was fixed on all other axes. The pulse was initiated from a trim condition and consisted of a control input of approximately 1/2 inch, which was then maintained for 1/2 to 1 second. The control was then returned to the trim position and held fixed until the motion had been established or recovery action was necessary.

14. The normal acceleration characteristics and the shapes of the individual parameters were evaluated against the requirements of Reference 12.

CONTROLLABILITY

15. The controllability was determined by the resulting maximum angular rates and accelerations per inch of control inputs. The angle at the end of 1 second after the control input was also evaluated for compliance with Reference 12. The step input was accomplished by rapidly displacing the control and maintaining this control position until the maximum rates and accelerations were reached. All other controls were held fixed during the step input. The control inputs were incrementally varied from 1/4 to approximately 1.5 inches. The longitudinal, lateral, and directional axes were evaluated.

APPENDIX IV

WEIGHT AND BALANCE

1. The test aircraft was weighed prior to installation of the test instrumentation. The weight and balance was conducted in a closed hangar using an electronic weighing kit. As received, the aircraft gross weight was 1148 pounds with a longitudinal C.G. location of 106.3 inches behind the reference line and a lateral C.G. location of .05 inch to the right of the center. The reference line is 100 inches forward of the rotor center line.

2. To verify the manufacturer's weight guarantees, the weight was corrected to the Chart A specifications (Reference 13). This included the removal of 5.9 pounds of oil and 2 jack fittings weighing 1.3 pounds. Weight was added to account for 2 cargo seats (5.3 pounds), 2 seat belts (2.4 pounds), unuseable fuel (1.4 pounds), and a flight manual (0.8 pounds). With these corrections, the empty weight was 1153 pounds at a longitudinal C.G. location of 106.0 inches. The manufacturer's empty weight guarantee is 1190 pounds with 37 pounds of fixed armor included. Including this fixed armor in the empty weight of the test helicopter, the 1190-pound guarantee is jet met. The Configuration II empty weight is 1517 pounds, which includes the XM-27E1 weapons system (237 pounds) plus 94 pounds of removable armor.

3. The Configuration I gross weight of 2290 pounds includes 400 pounds for a crew of two, 94 pounds of removable armor, 200 pounds of mission equipment, and sufficient fuel for a range of 260 nautical air miles (conditions as specified in References 1 and 2). Added to the empty gross weight of 1190 pounds, this allows 406 pounds for fuel. The Configuration II gross weight at 2500 pounds includes 400 pounds for a crew of two, 200 pounds of mission equipment, and sufficient fuel for a range of 230 nautical air miles (conditions as specified in References 1 and 2). Added to the empty weight of 1517 pounds, this allows for 383 pounds of fuel.

1190
200
94
406
2290

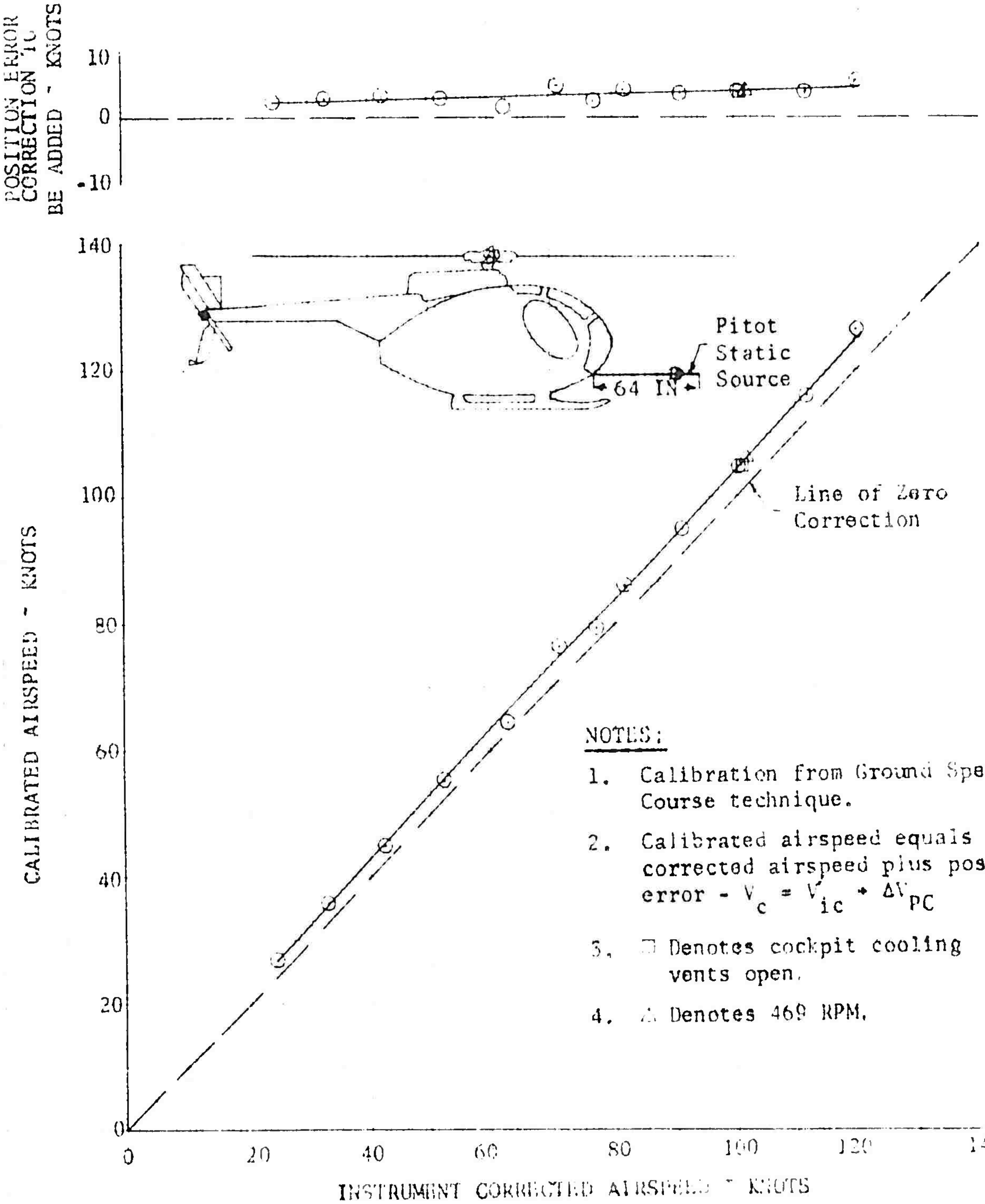
4. The aircraft was weighed after installation of test instrumentation, and the basic weight (full oil, no fuel) was 1510 pounds with a longitudinal C.G. location of 104.3 inches.

APPENDIX V

GRAPHICAL TEST DATA

FIGURE NO. 1
AIRSPEED CALIBRATION
OH-6A S/N 65-12927
CLEAN CONFIGURATION
BOOM SYSTEM

DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION LONG IN	C.G. LOCATION LAT	FLIGHT CONDITION
2320	2260	483	100.3 (Mid)	0 (Mid)	Level Flight



NOTES:

1. Calibration from Ground Speed Course technique.
2. Calibrated airspeed equals corrected airspeed plus position error - $V_c = V_{ic} + \Delta V_{PC}$
3. □ Denotes cockpit cooling vents open.
4. △ Denotes 469 RPM.

FIGURE NO. 2
AIRSPEED CALIBRATION
OH-6A S/N 65-12927
CLEAN CONFIGURATION
STANDARD SYSTEM

DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
2320	2260	483	100.3 (Mid)	0 (mid)	Level Flight

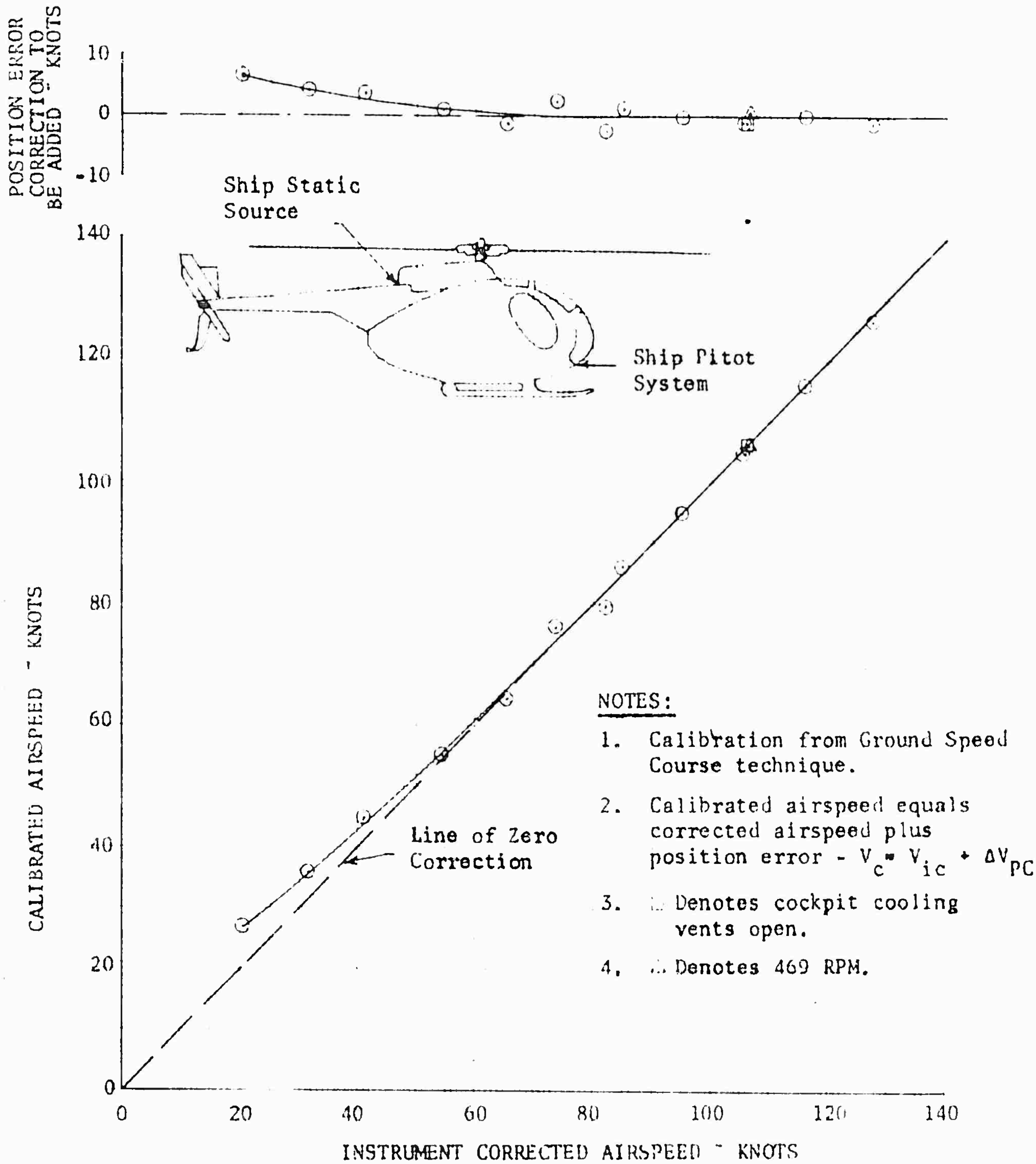
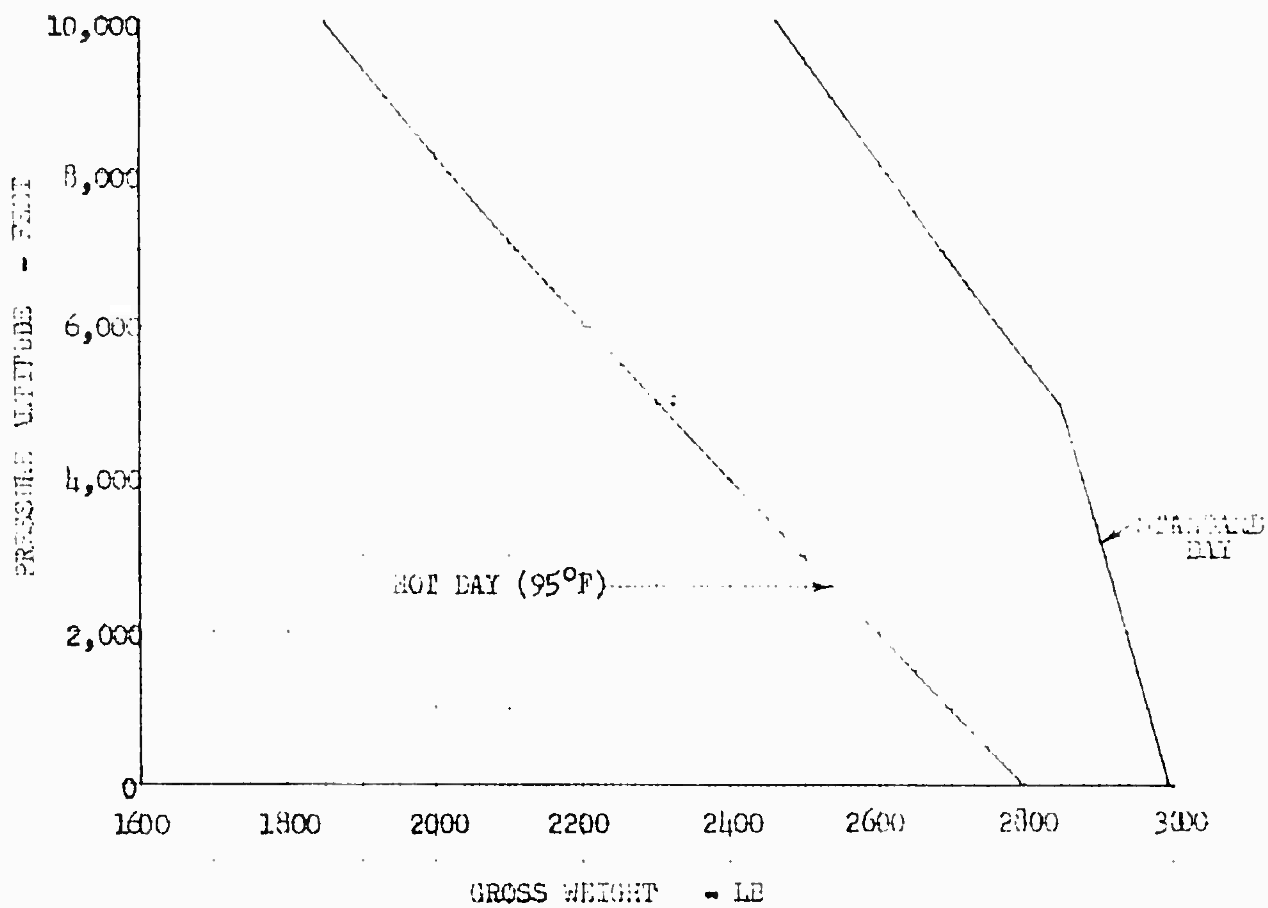


FIGURE NO. 3
SUMMARY HOVERING PLOT
OH-6A S/N 65-12927
IN GROUND EFFECT
CLEAN CONFIGURATION
4 FT SKID HEIGHT
TAKE OFF POWER

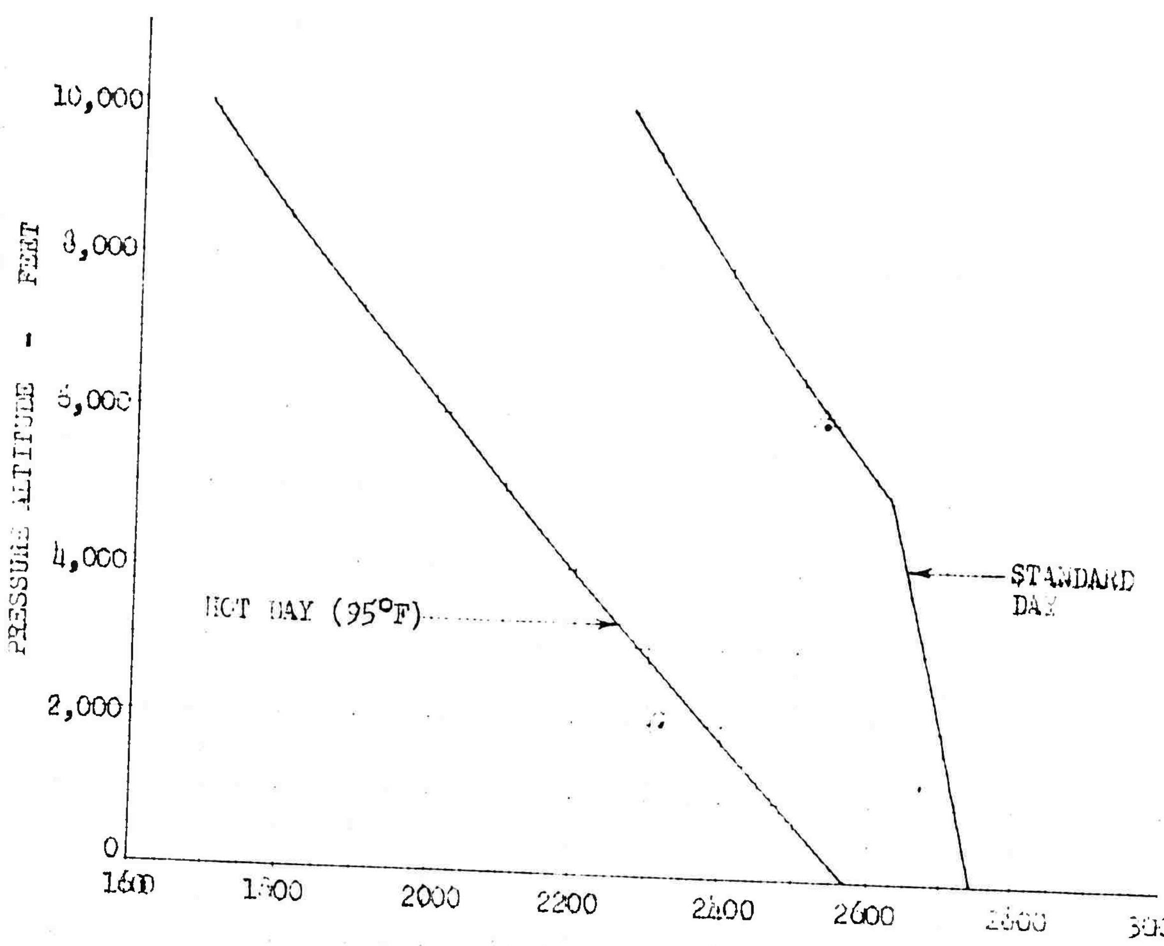
NOTE: Curves derived from figure 5.



FOR OFFICIAL USE ONLY

FIGURE NO. 4
SUMMARY HOVERING PLOT
OH-6A S/N 65-12927
OUT OF GROUND EFFECT
CLEAN CONFIGURATION
TAKE OFF POWER
55 FT SKID HEIGHT

NOTE: Curves derived from Figure 6.



FOR OFFICIAL USE ONLY

FIGURE NO. 5
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION
SKID HEIGHT = 4 FT

SYM	DENSITY ALTITUDE FT
○	5820
□	670

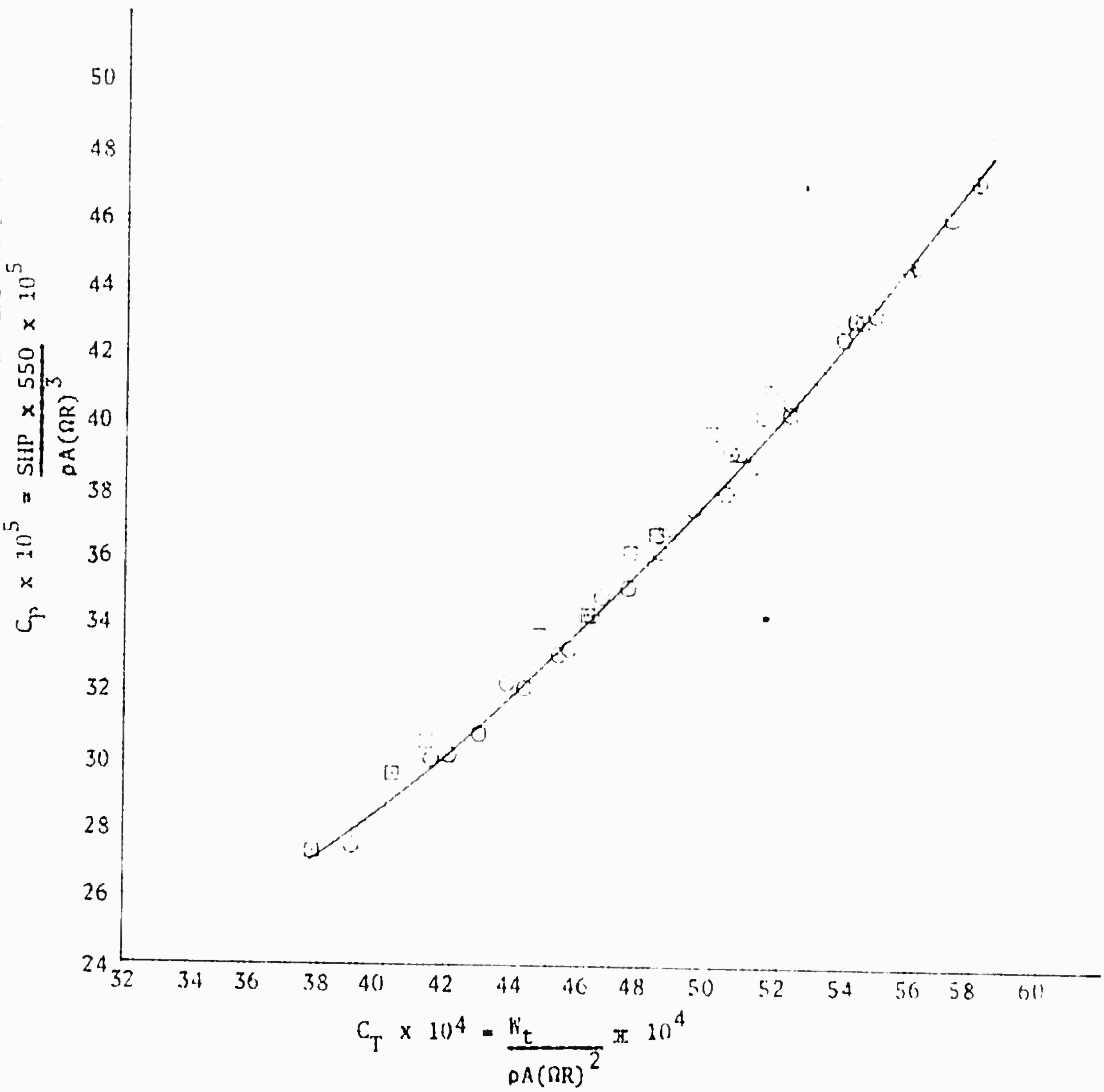


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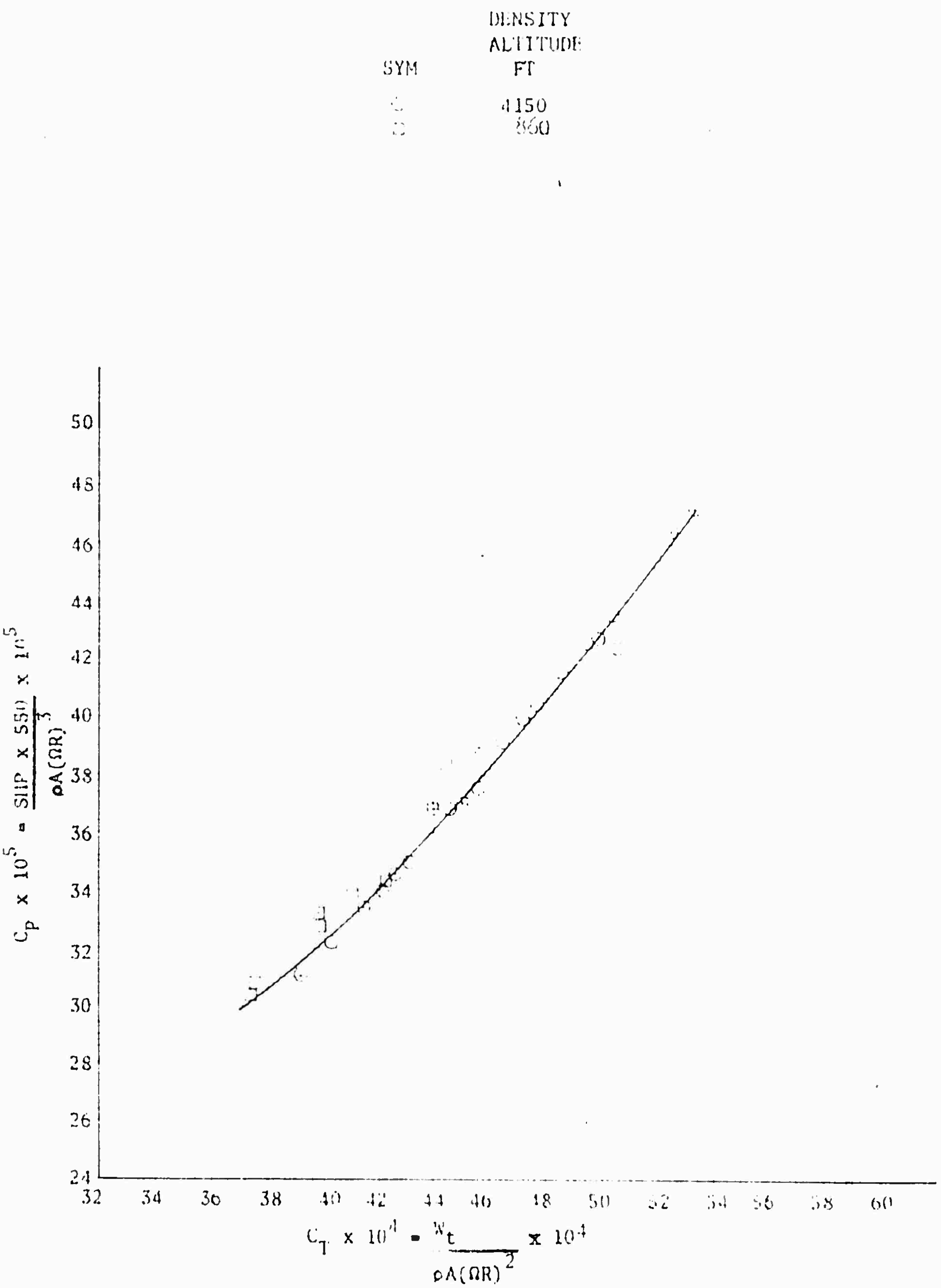
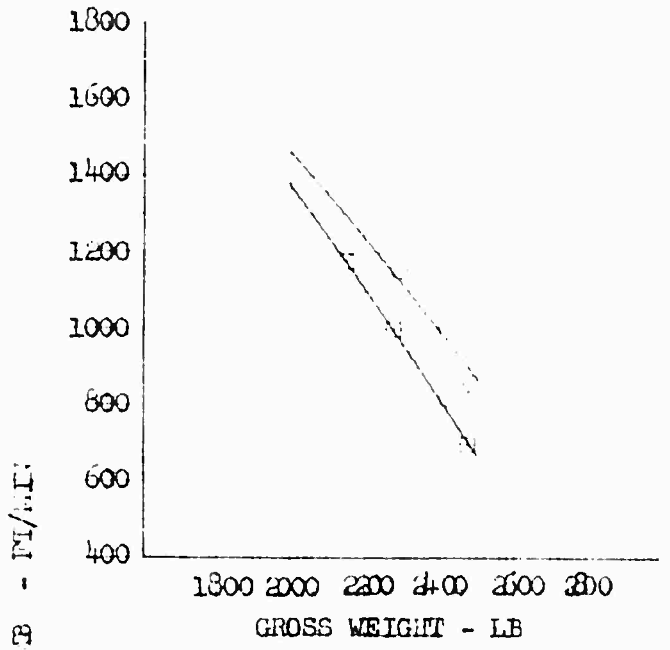
FIGURE NO. 6
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION
SKID HEIGHT = 35 FT

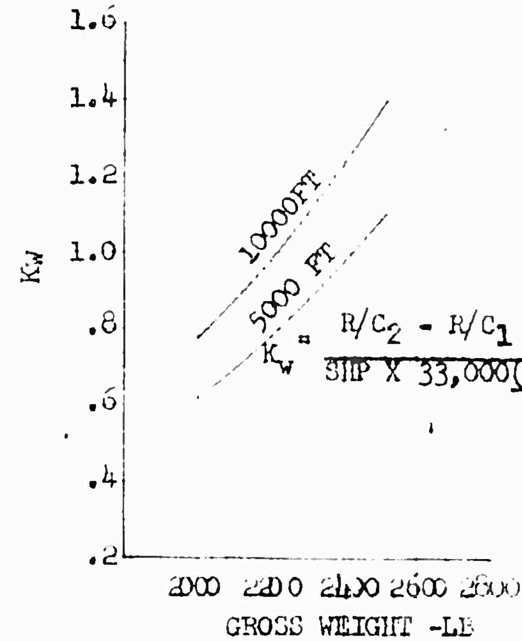


FOR OFFICIAL USE ONLY

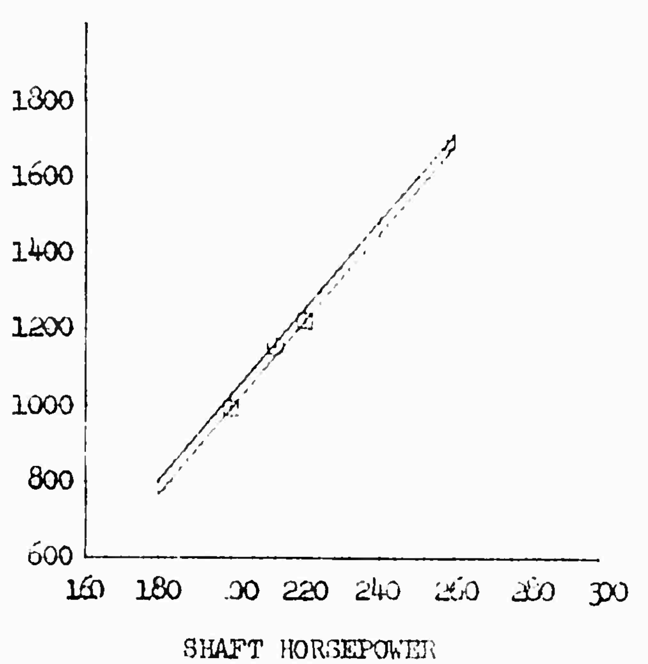
FIGURE NO. 7
VARIATION IN RATE OF CLIMB
AS A FUNCTION OF GROSS WEIGHT
AND SHAFT HORSEPOWER
OH-6A S/N 65-12927

SYM	DENSITY ALTITUDE FT	SHP	ROTOR SPEED RPM
	5000	212	483
	10000	200	483





$$K_W = \frac{R/C_2 - R/C_1}{SHP \times 33{,}000\left(\frac{1}{GW_2} - \frac{1}{GW_1}\right)}$$



SYM	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM
	5000	2290	483
	10000	2290	483

K_p = .810

$$K_p = \frac{\Delta R/C}{\Delta SHP} \times \frac{GW}{33000}$$

FIGURE NO. 8
CLIMB PERFORMANCE
OH-6A S/N 65-12927

CLEAN CONFIGURATION
TAKEOFF POWER
STANDARD DAY
ROTOR SPEED = 483 RPM
C.G. LOCATION = 100 (MID)

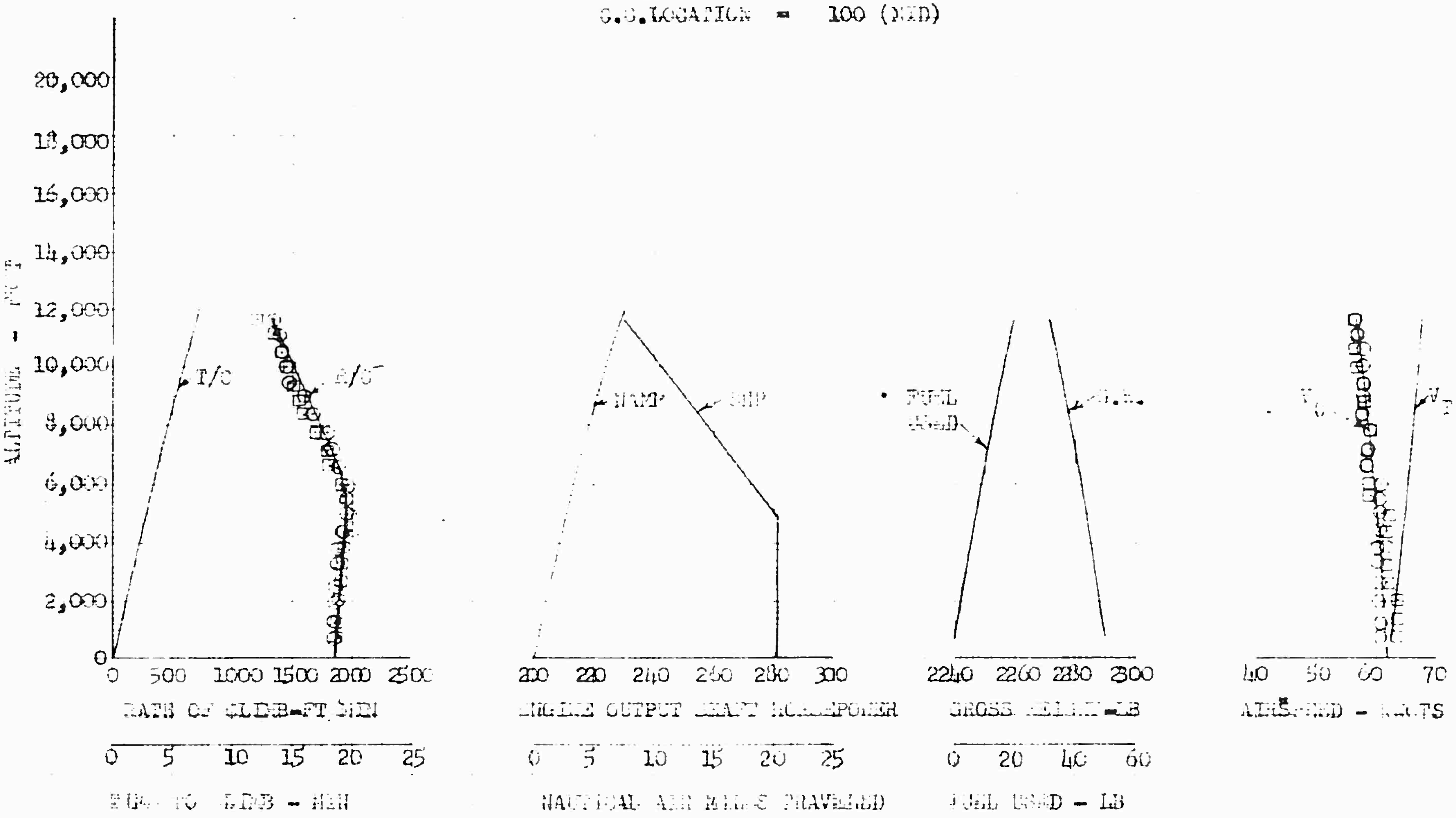


FIGURE NO. 9
CLIMB PERFORMANCE
OH-6A S/N 65-12927

CLEAN CONFIGURATION
MAX CONTINUOUS POWER
STANDARD DAY
ROTOR SPEED = 483 RPM
C.G. LOCATION = 100(MID)

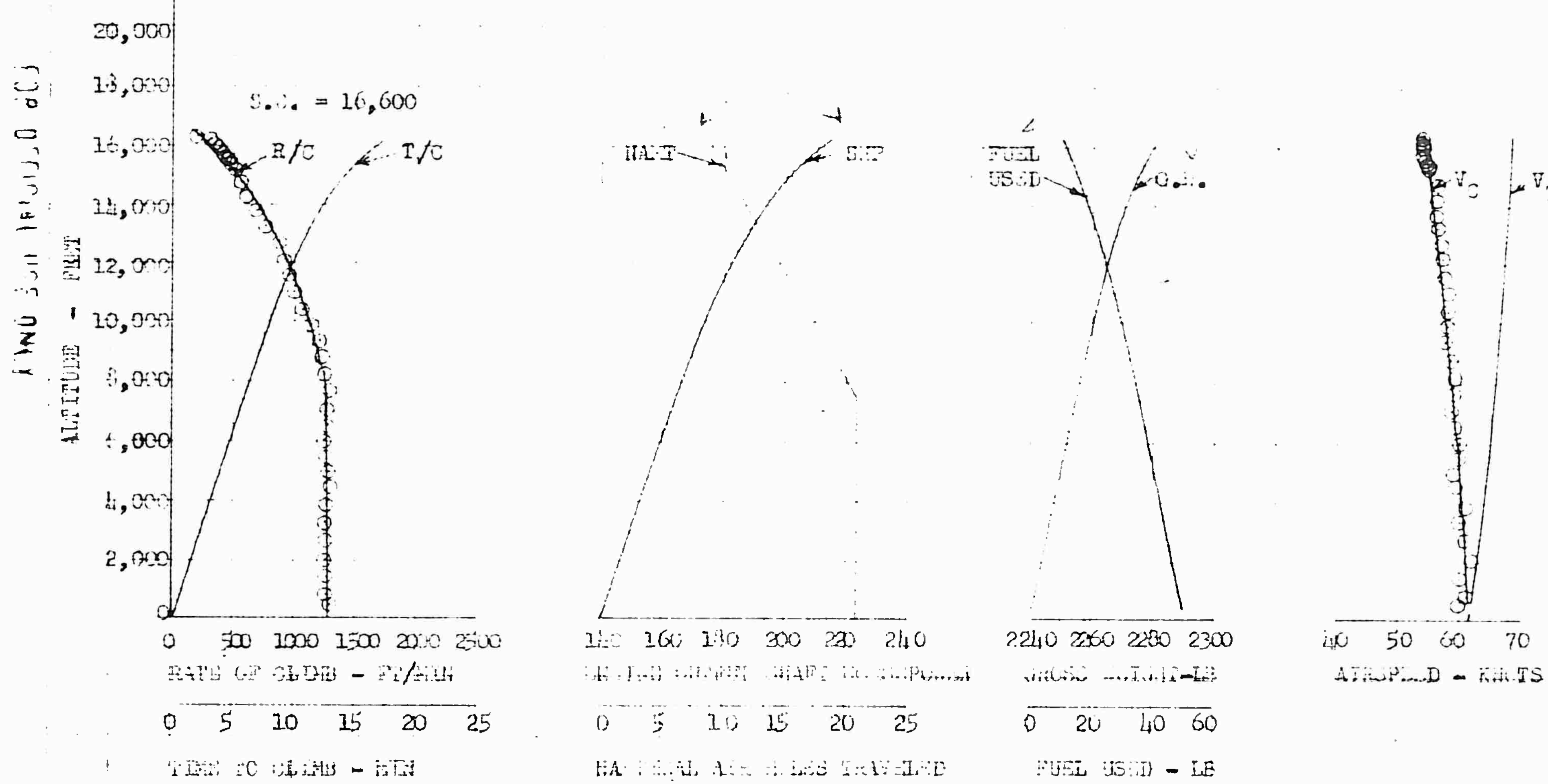


FIGURE NO. 10
RANGE PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION
ROTOR SPEED = 483 RPM
C.G.LOCATION = 97 (FWD)

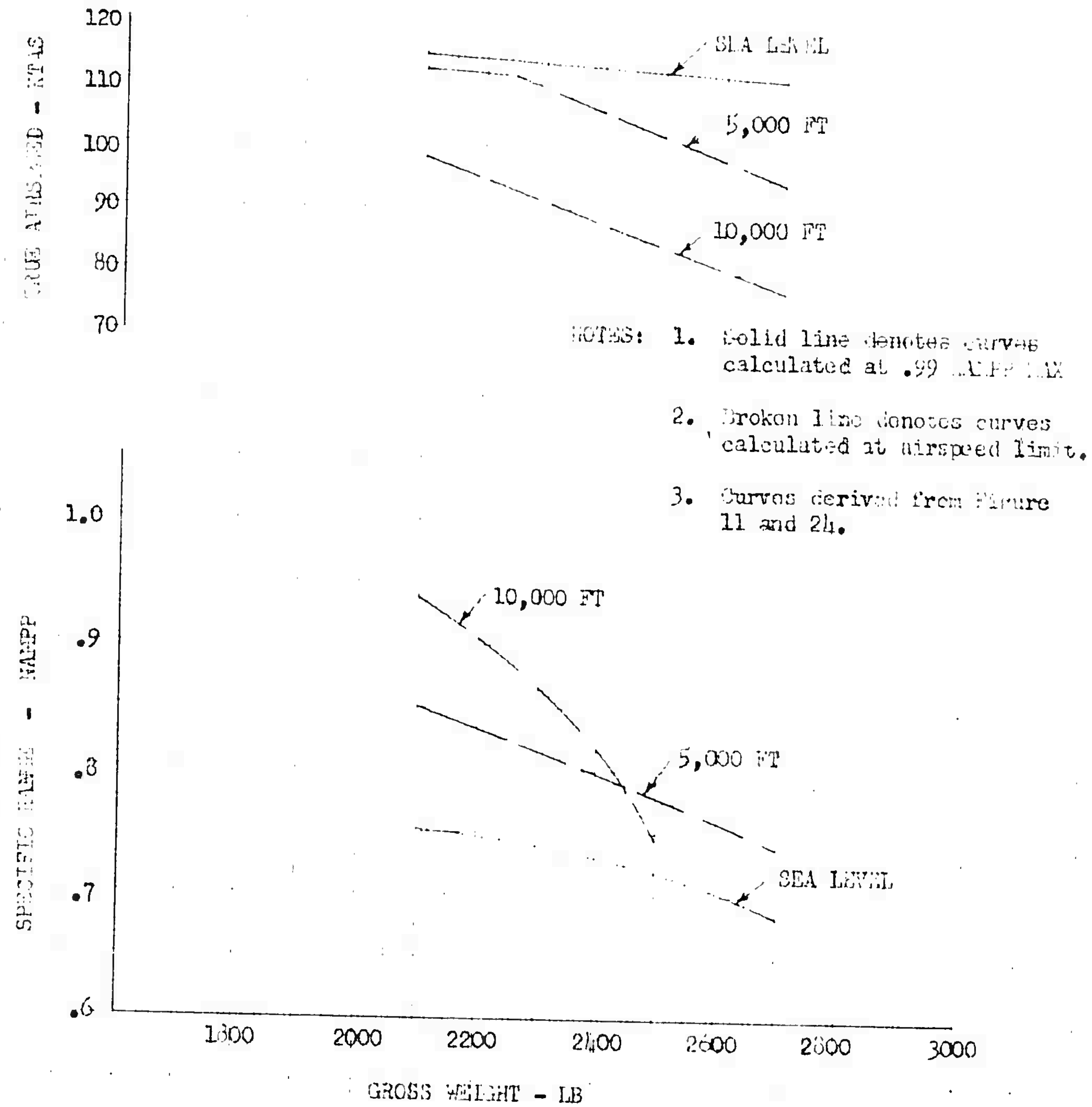


FIGURE NO. 11 (a)
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

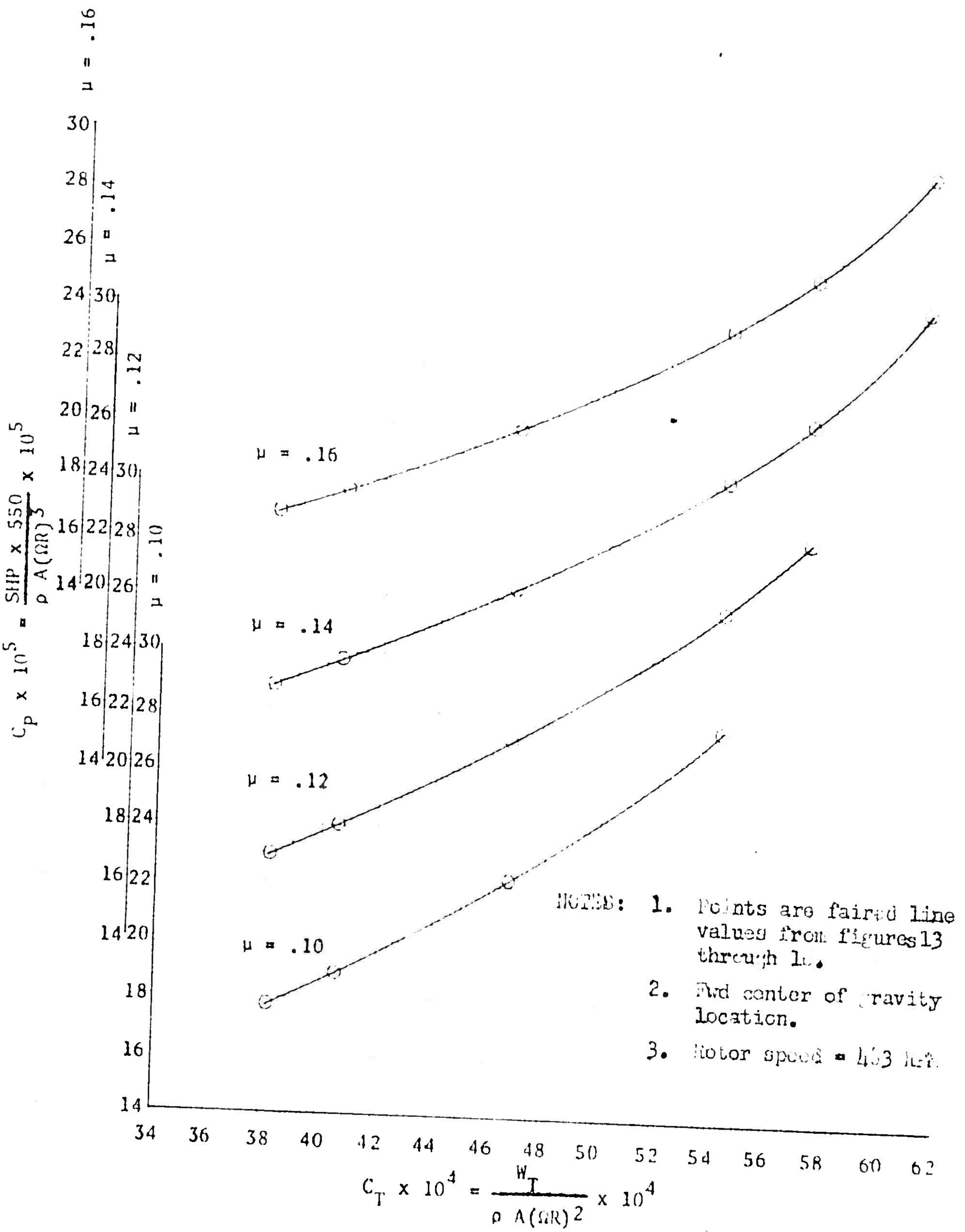


FIGURE NO. 11 (b)
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

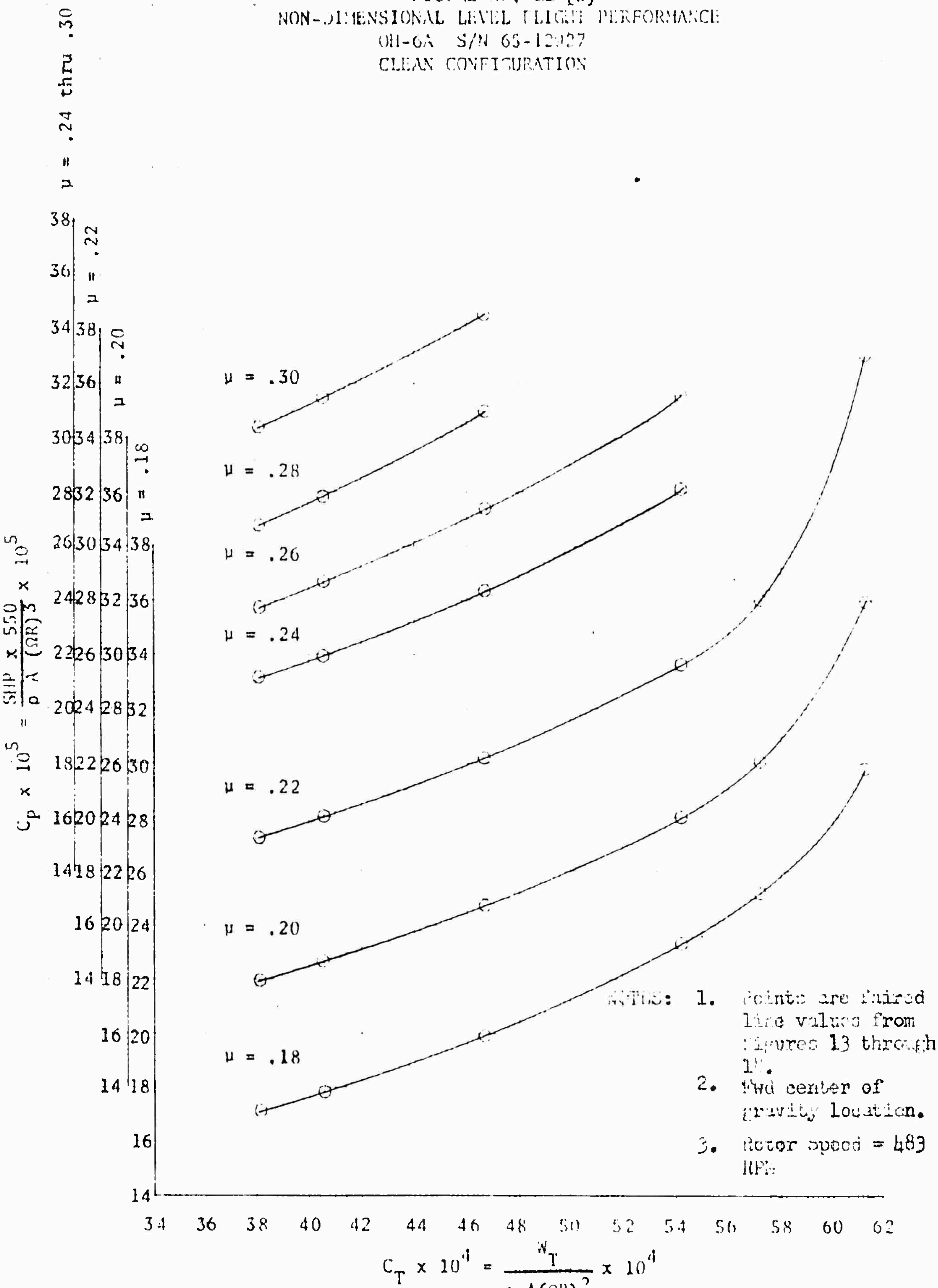


FIGURE NO. 12
EFFECT OF CENTER OF GRAVITY LOCATION ON LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

SYM	$C_T \times 10^{-4}$	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
○	46.72	5270	2290	483	97.0(fwd)	0(mid)	Level Flight
□	46.55	5060	2300	483	100.0(mid)	0(mid)	Level Flight
△	46.41	5550	2290	483	104.0(aft)	1.4(lt)	Level Flight

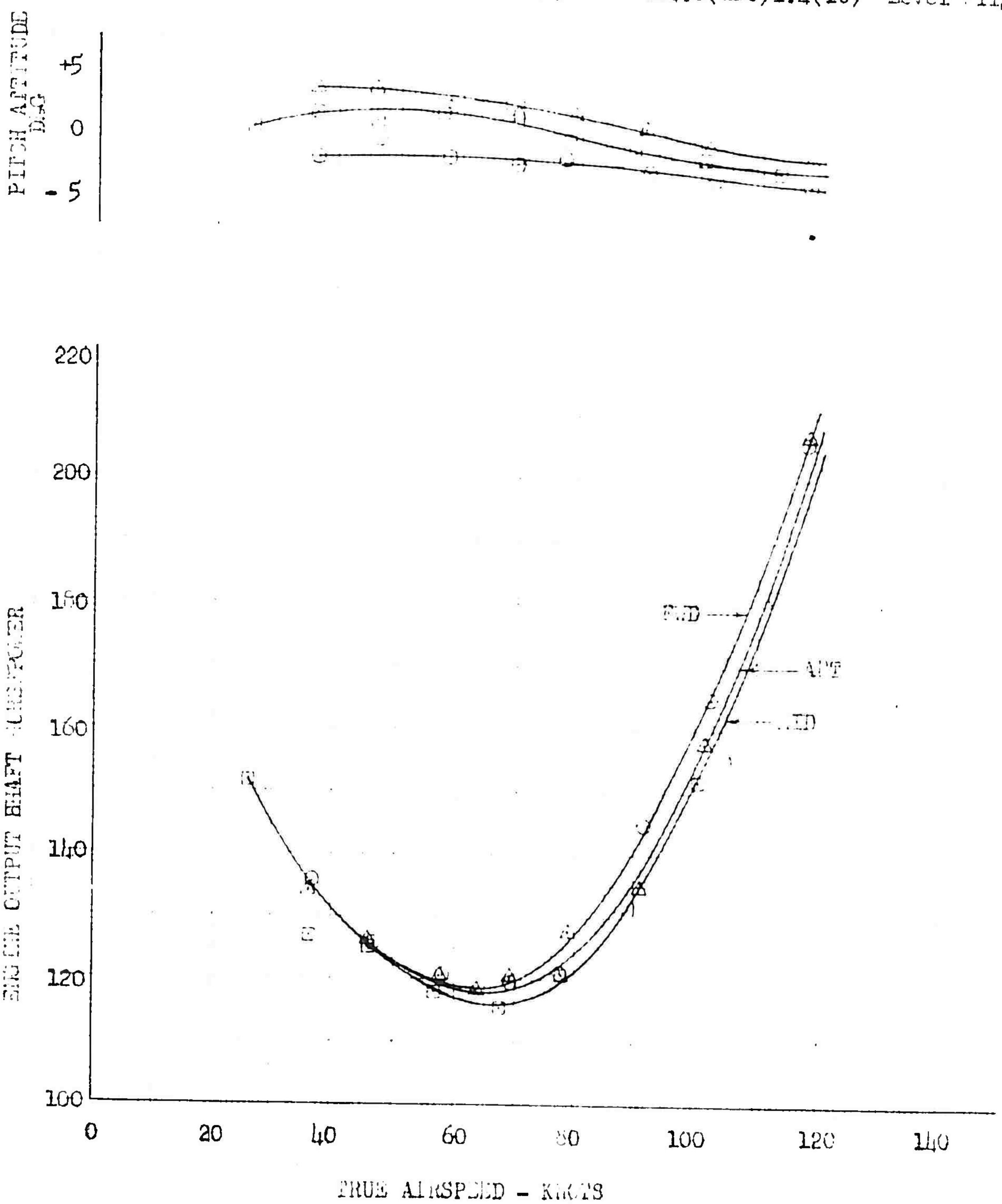


FIGURE NO. 13
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

GROSS WEIGHT	=	2190 LB
DENSITY ALTITUDE	=	Sea Level
ROTOR SPEED	=	483 RPM
C.G. LOCATION	=	Station 97.0 (Fwd)
C_T	=	.003813

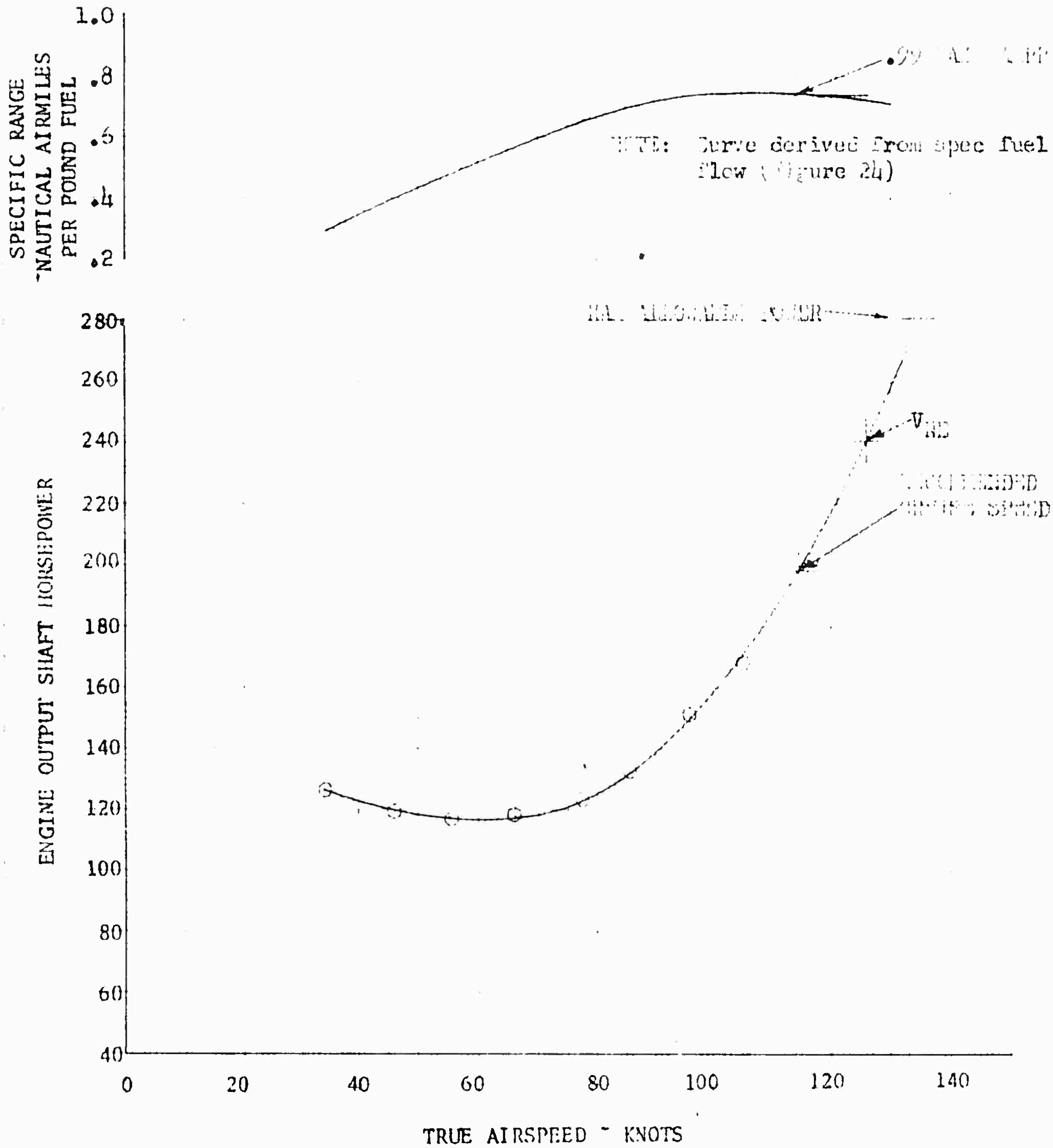


FIGURE NO. 14
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

GROSS WEIGHT = 2330 LB
DENSITY ALTITUDE = Sea Level
ROTOR SPEED = 483 RPM
C.G. LOCATION = Station 97.0 (Fwd)
C_T = .004054

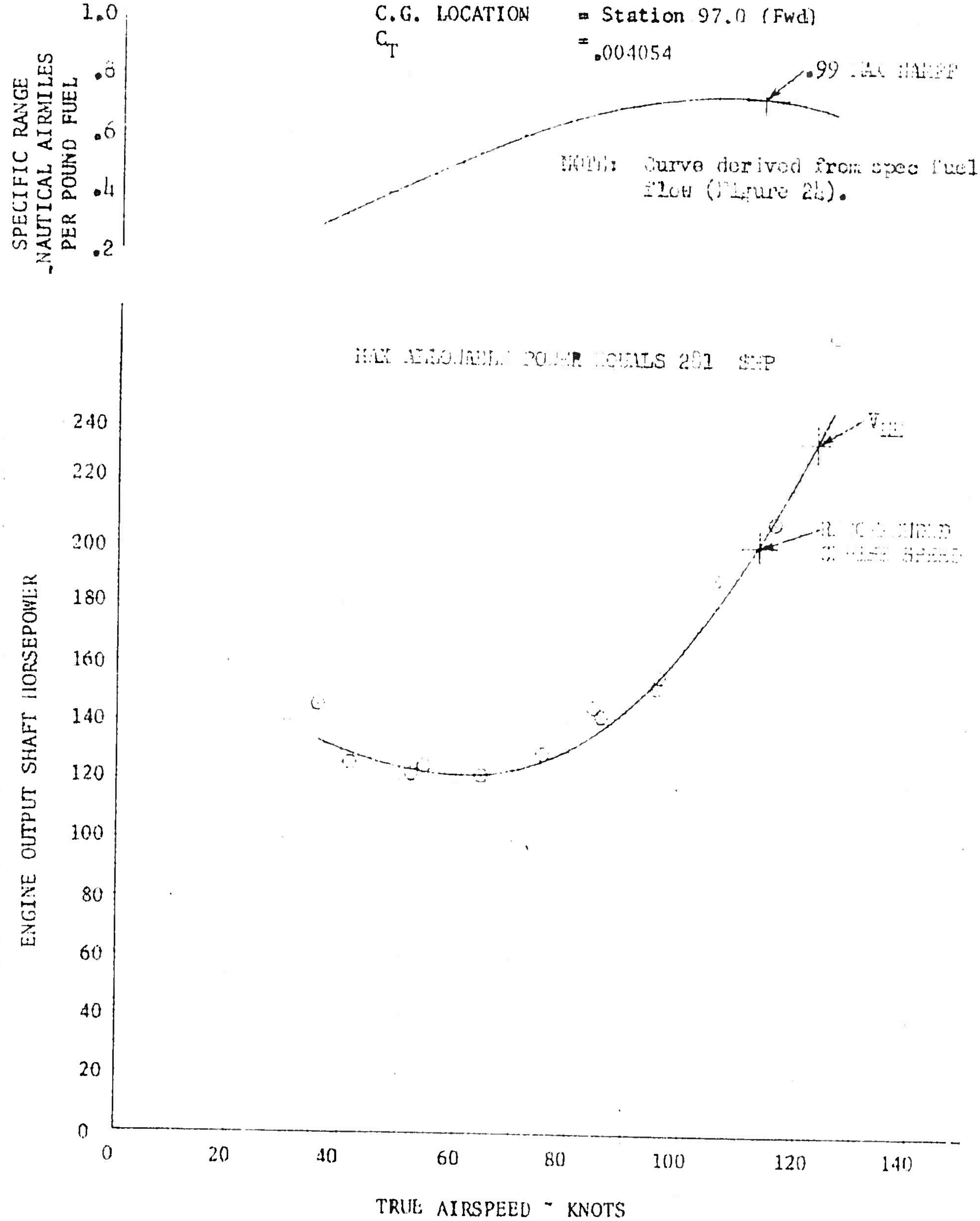


FIGURE NO. 15
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

GROSS WEIGHT = 2310 LB
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 483 RPM
C.G. LOCATION = Station 97.0 (Fwd)
C_T = .004672

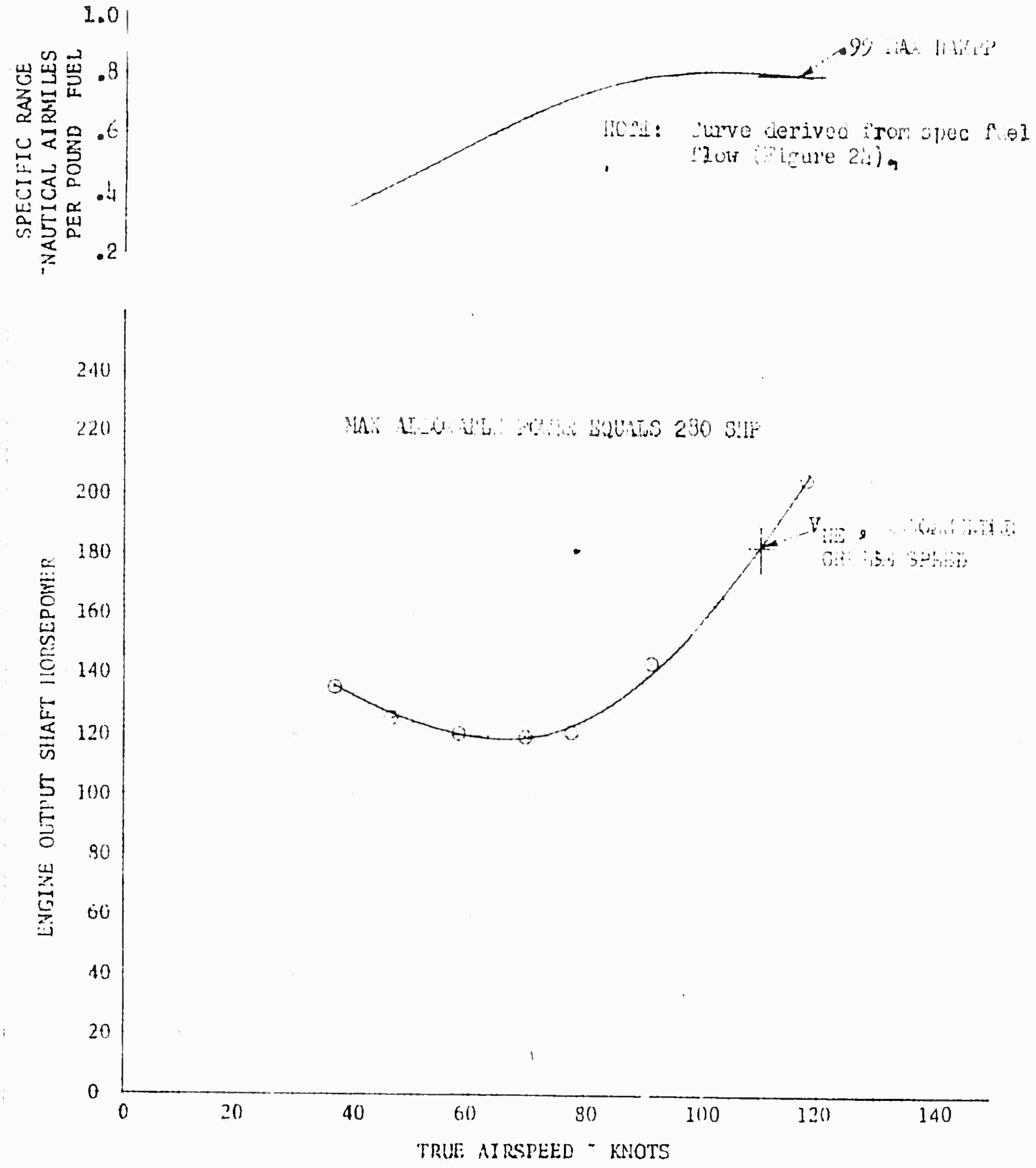


FIGURE NO. 16
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

GROSS WEIGHT	=	2300 LB
DENSITY ALTITUDE	=	10,000 FT
ROTOR SPEED	=	483 RPM
C.G. LOCATION	=	Station 97.1(fwd)
C_T	=	.005422

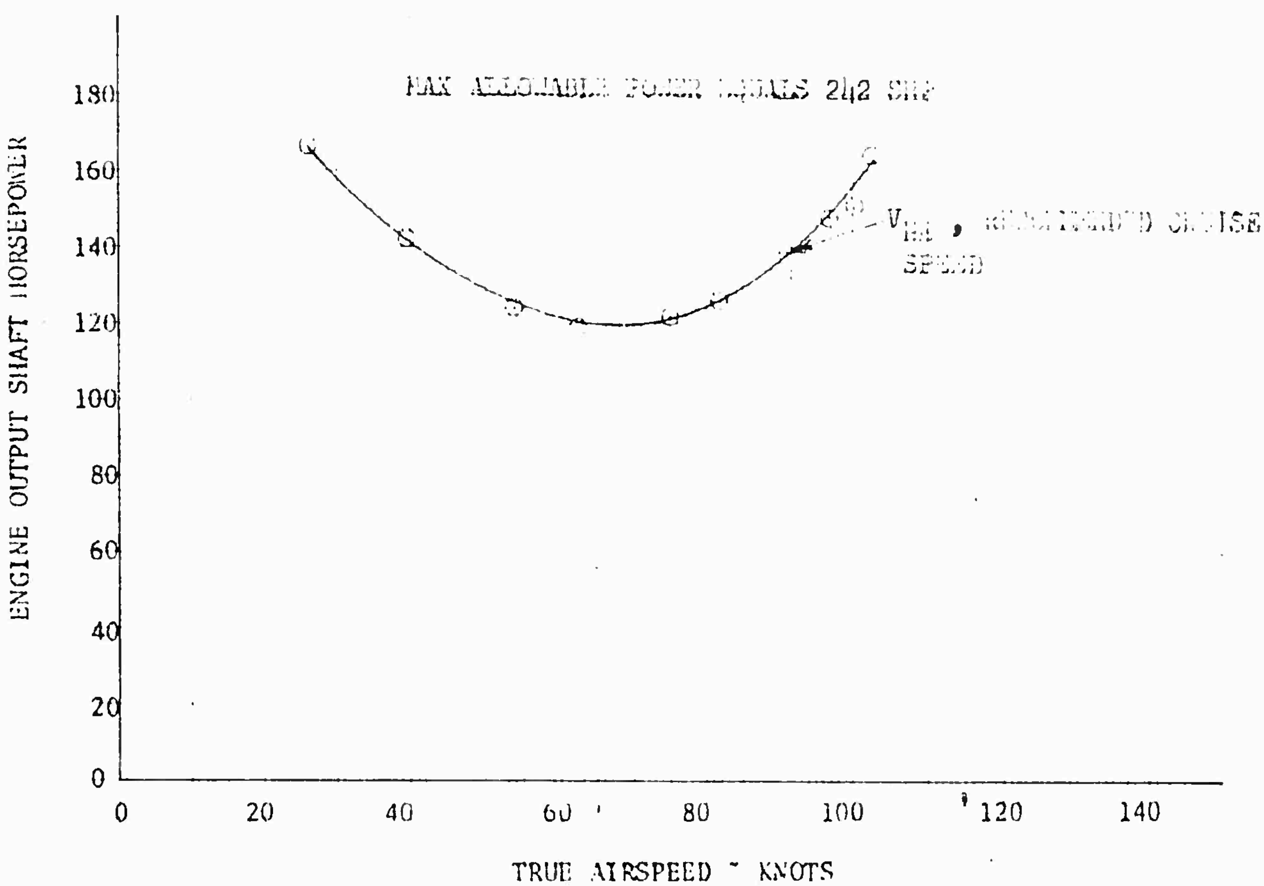


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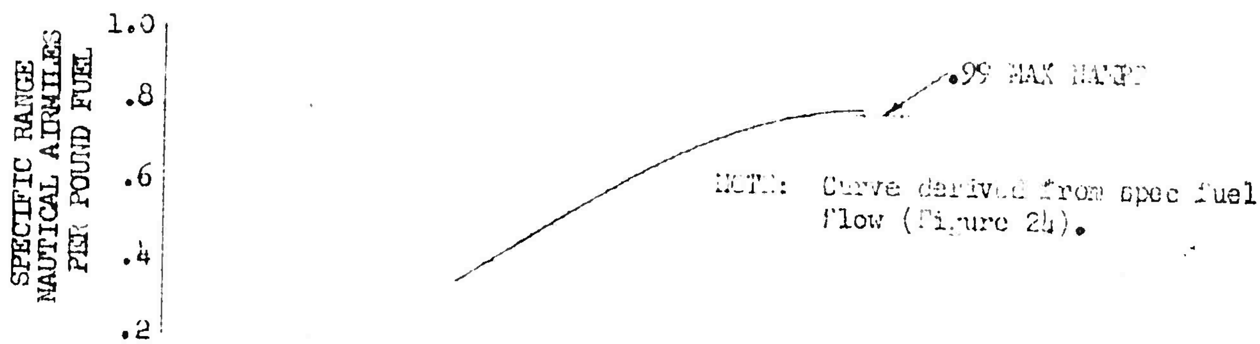
FIGURE NO. 17
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

GROSS WEIGHT	2430 LB
DENSITY ALTITUDE	10,000 FT
ROTOR SPEED	483 RPM
C.G. LOCATION	Station 97.0(Fwd)
C_T	.005721



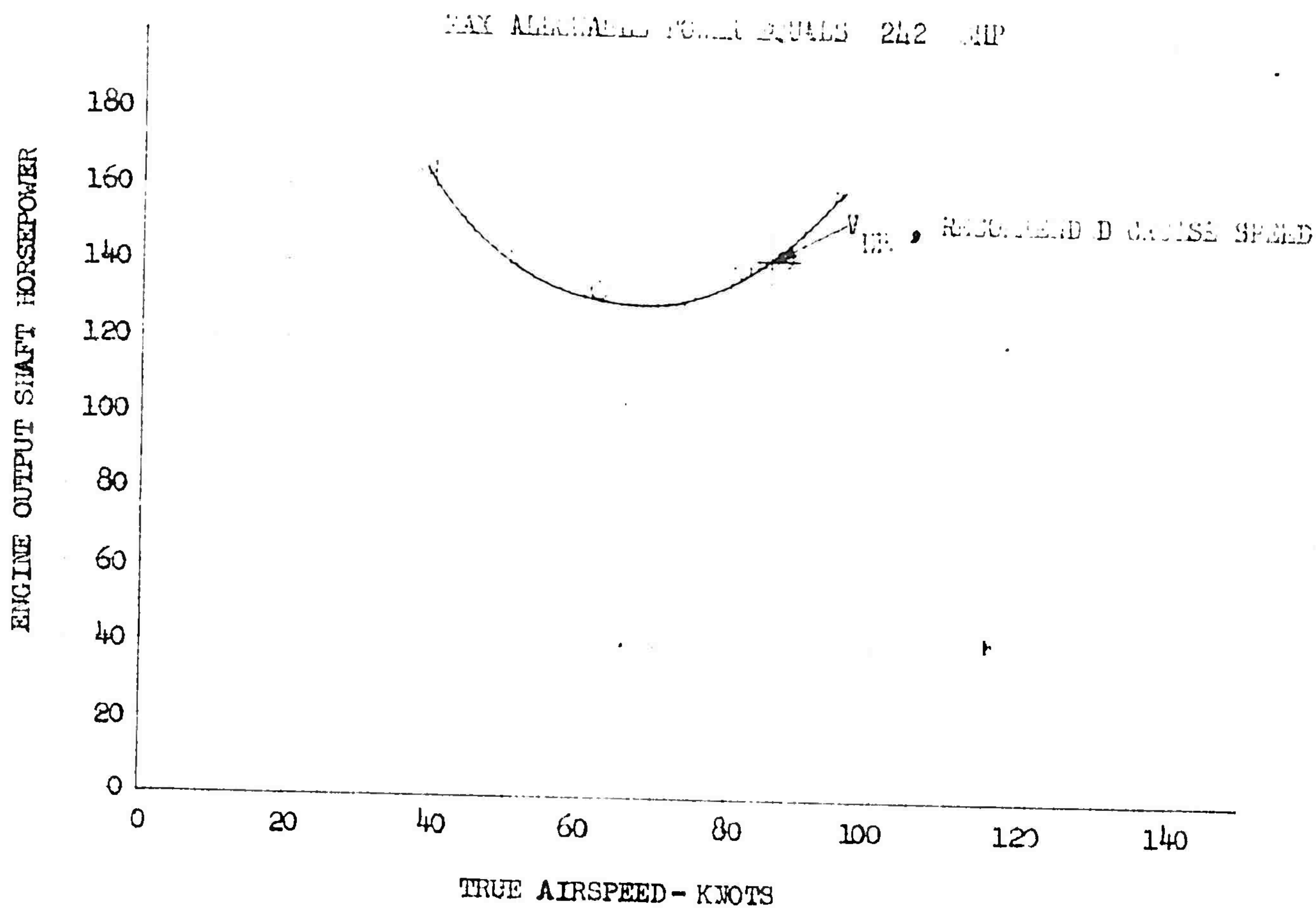


FIGURE NO. 18
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

GROSS WEIGHT	2290 LB
DENSITY ALTITUDE	14000 FT
ROTOR SPEED	483 RPM
C.G. LOCATION	Station 97.0 (Fwd)
C_T	.006128

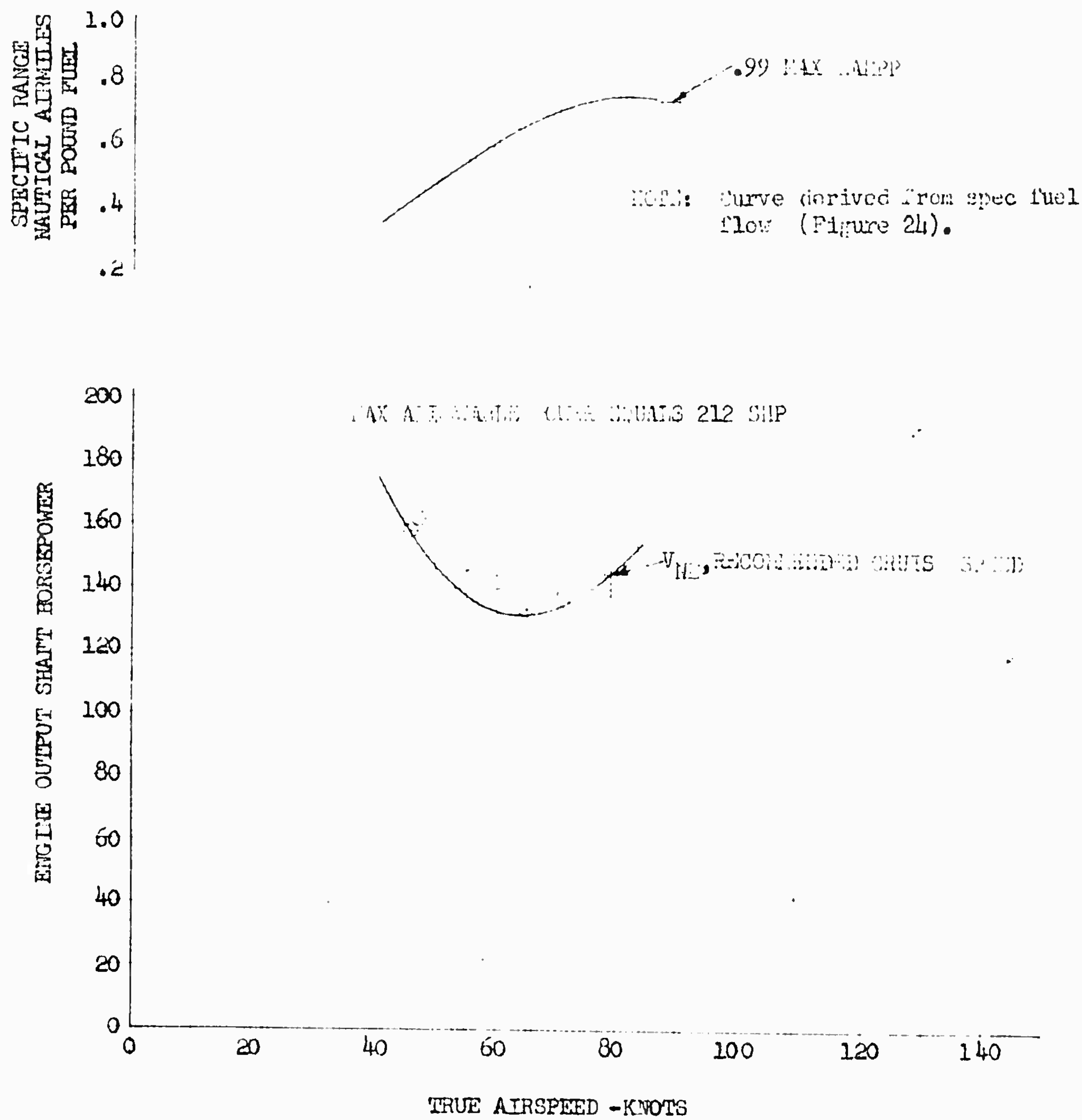


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FIGURE NO. 19
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
ARMED CONFIGURATION
XM-27-E1 IN STOWED POSITION

GROSS WEIGHT	=	2660 LB
DENSITY ALTITUDE	=	Sea Level
ROTOR SPEED	=	483 RPM
C.G. LOCATION	=	Longitudinal Station 97.1 (Fwd)
		Lateral Station 1.5 (LT)
C_T	=	.004628

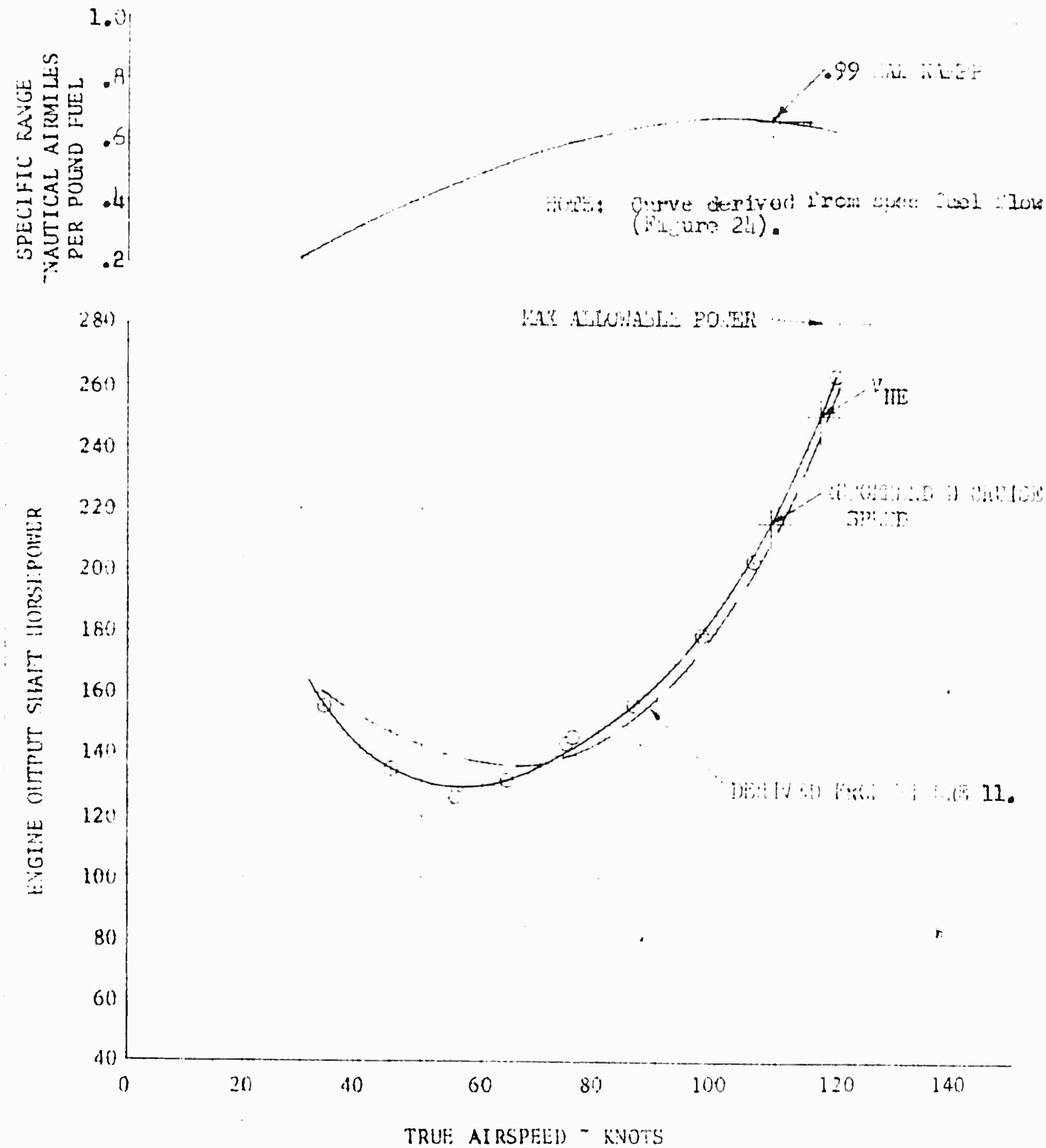

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FIGURE NO. 20
LEVEL FLIGHT PERFORMANCE
OH-6A S/N 65-12927
ARMED CONFIGURATION
XM-27-E1 IN STOWED POSITION

GROSS WEIGHT = 2530 LB
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 483 RPM
C.G. LOCATION = Longitudinal Station 97.1 (Fwd)
Lateral Station 1.5 (LT)
C_T = .005112

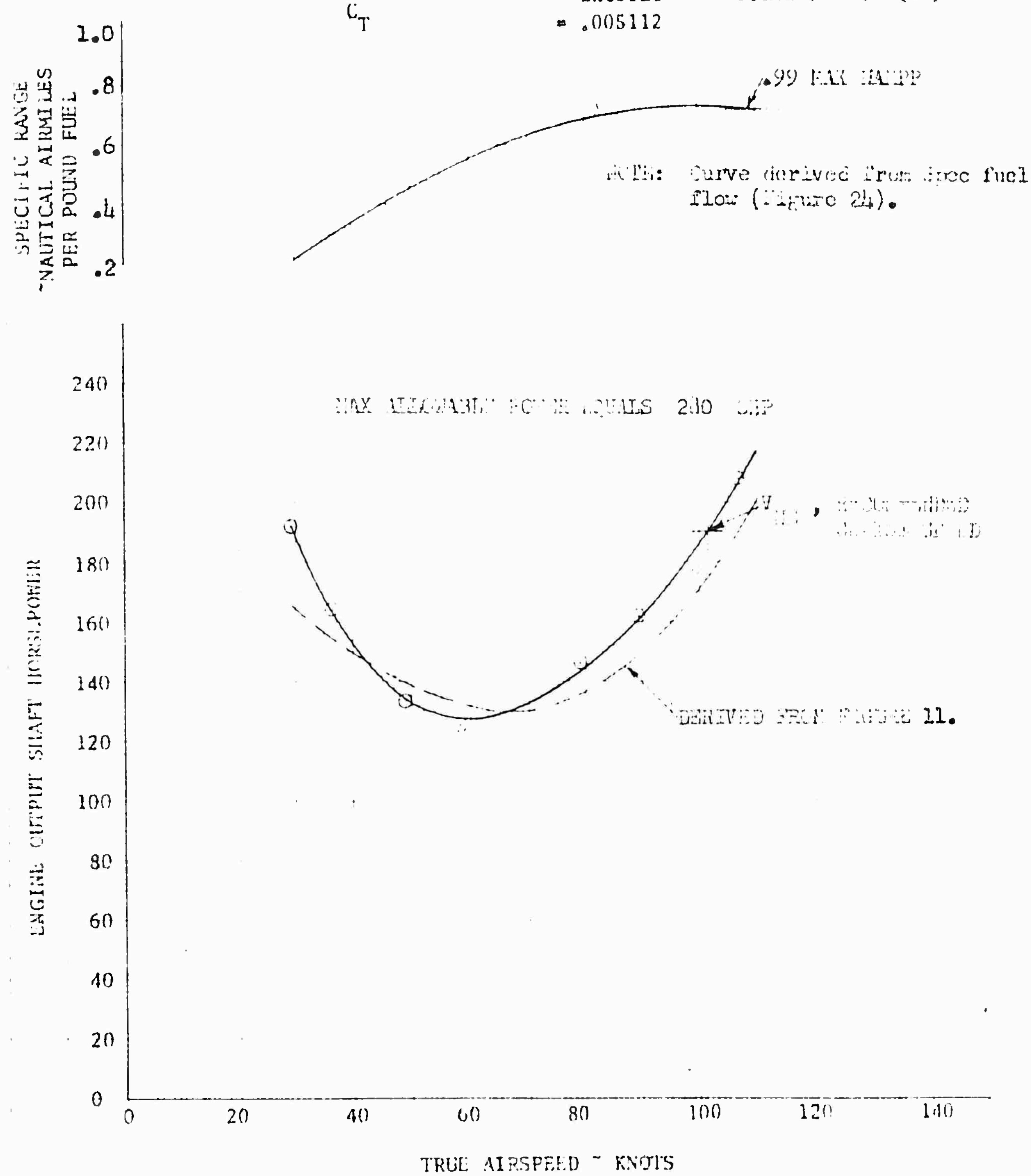


FIGURE NO. 21

SHAFT HORSEPOWER AVAILABLE
OH-6A S/N 65-12927

MAXIMUM CONTINUOUS POWER

$T_{T_5} = 693°C$

NOTE: 1. Based on compressor inlet condition as defined in Figure 23 at zero airspeed.

2. Shaft horsepower derived from Engine Model Specification 580-F (Ref 3).

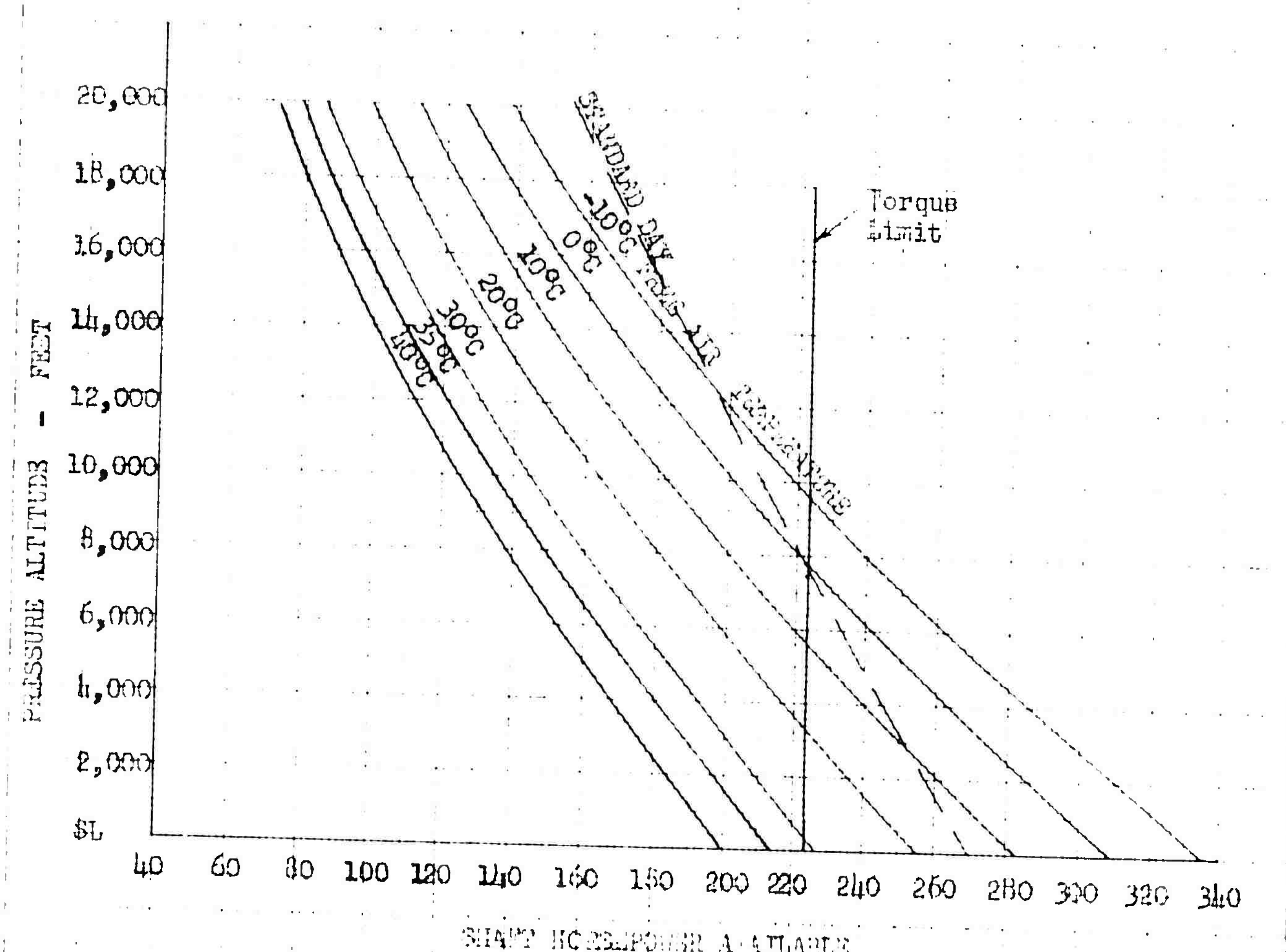


FIGURE NO. 22
SHAFT HORSEPOWER AVAILABLE
OH-6A S/N 65-12927

TAKEOFF POWER
T_{T_5} = 749°C

NOTE: 1. Based on compressor inlet condition as defined in figure 23 at zero airspeed.

2. Shaft horsepower derived from Engine Model Specification 580-F (Ref 3).

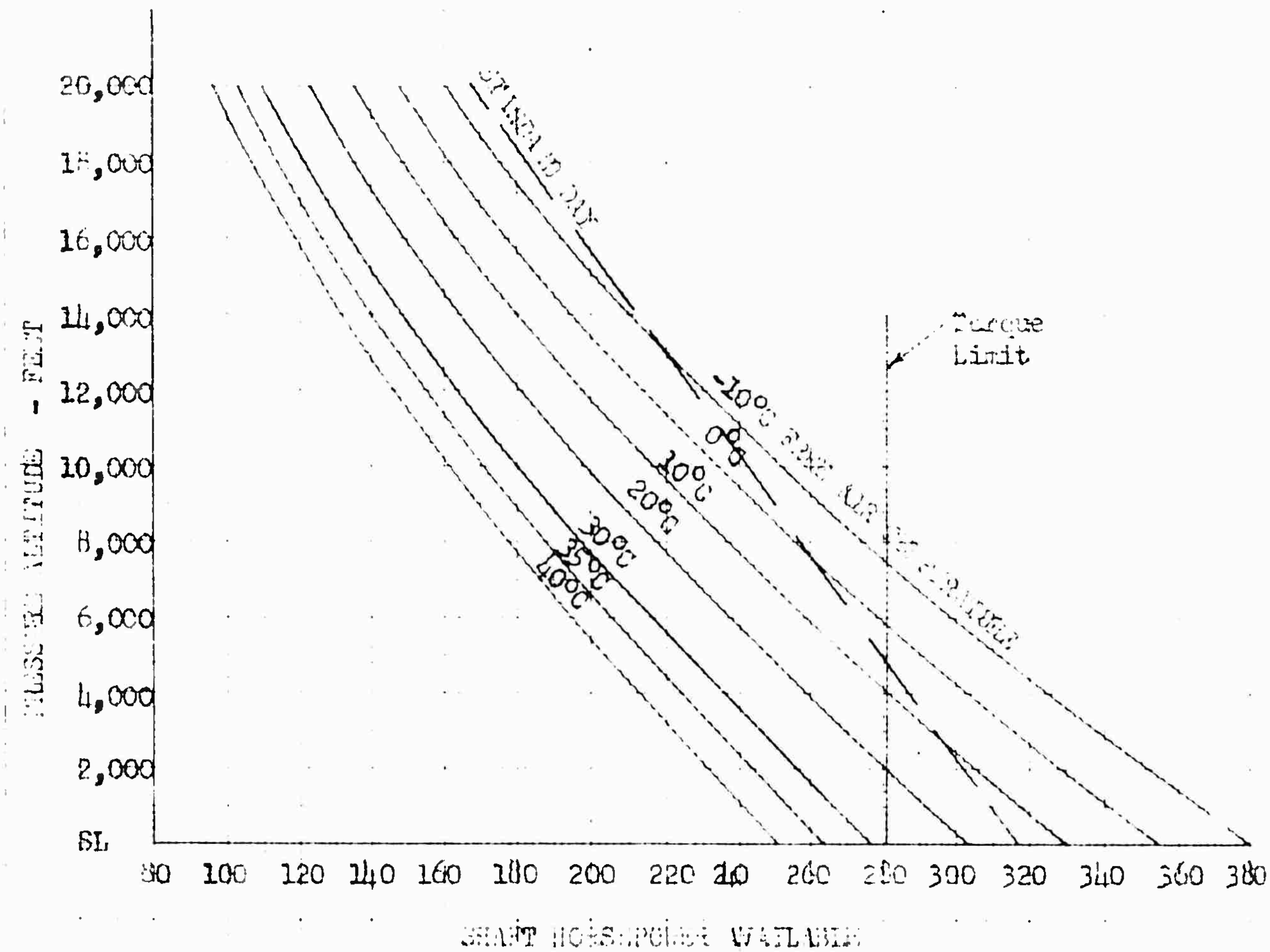


FIGURE NO. 23
INLET PERFORMANCE
OH-6A S/N 65-12927

SYM	
○	SEA LEVEL
□	5000 FT

NOTES:
1. P_{T_2} is compressor inlet total pressure.
2. T_{T_2} is compressor inlet total temperature.
3. Points were obtained during stabilized flight. Reference figures 14 through 15.

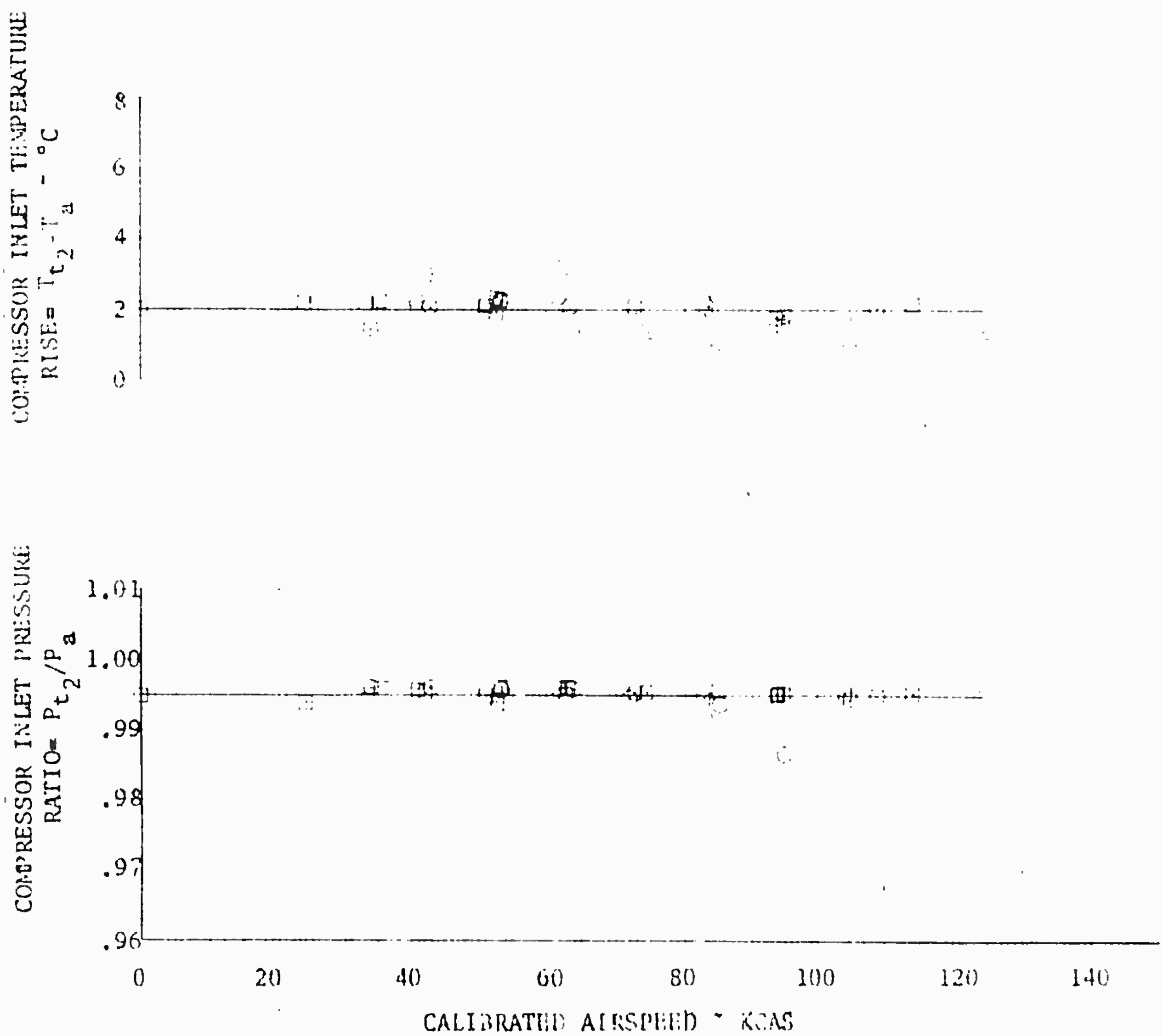


FIGURE NO. 24
ENGINE CHARACTERISTICS
OH-6A S/N 65-12927
T63-A-5A

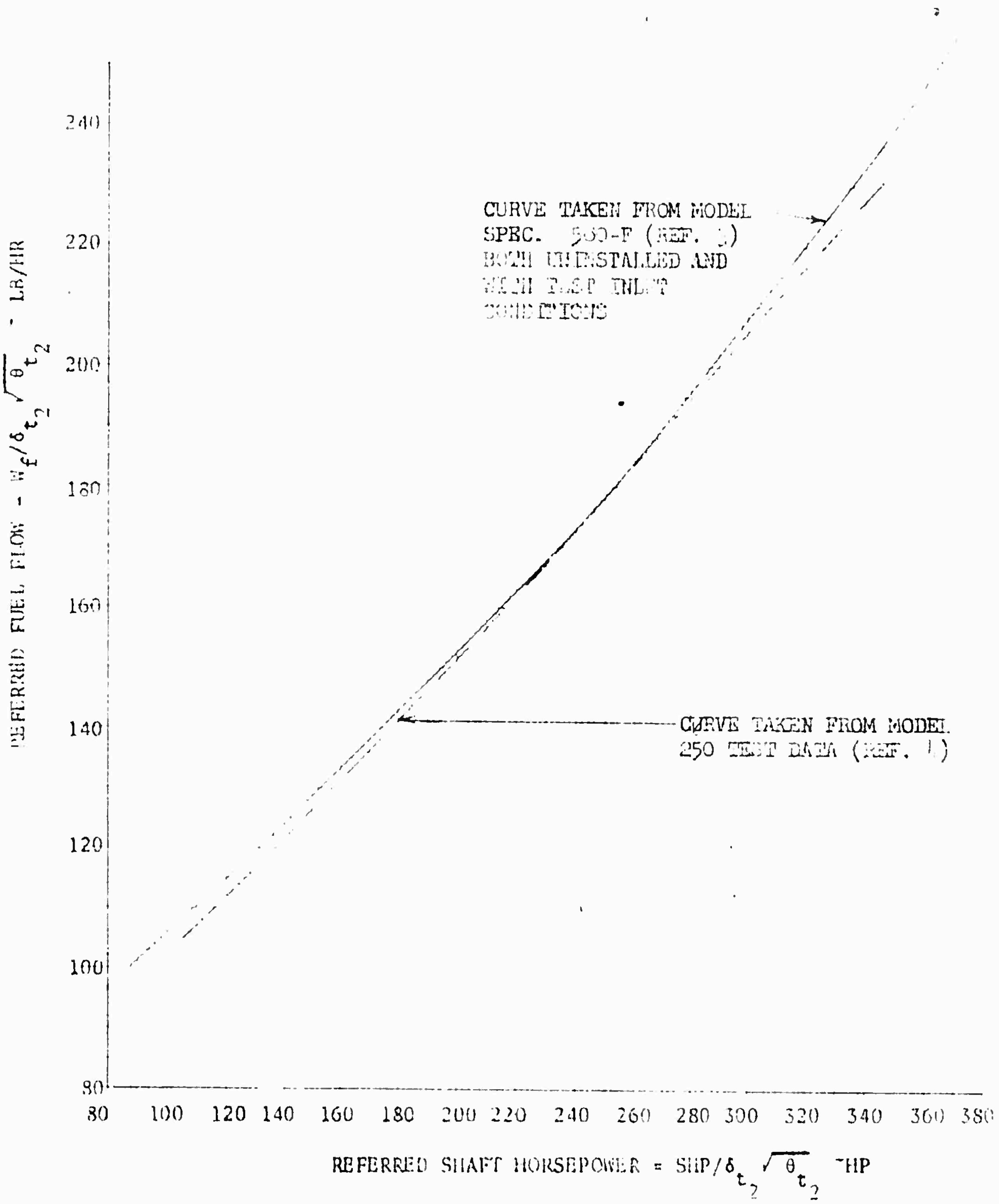


FIGURE NO. 25
ENGINE CHARACTERISTICS
OH-6A S/N 65-12927
T63-A-5A

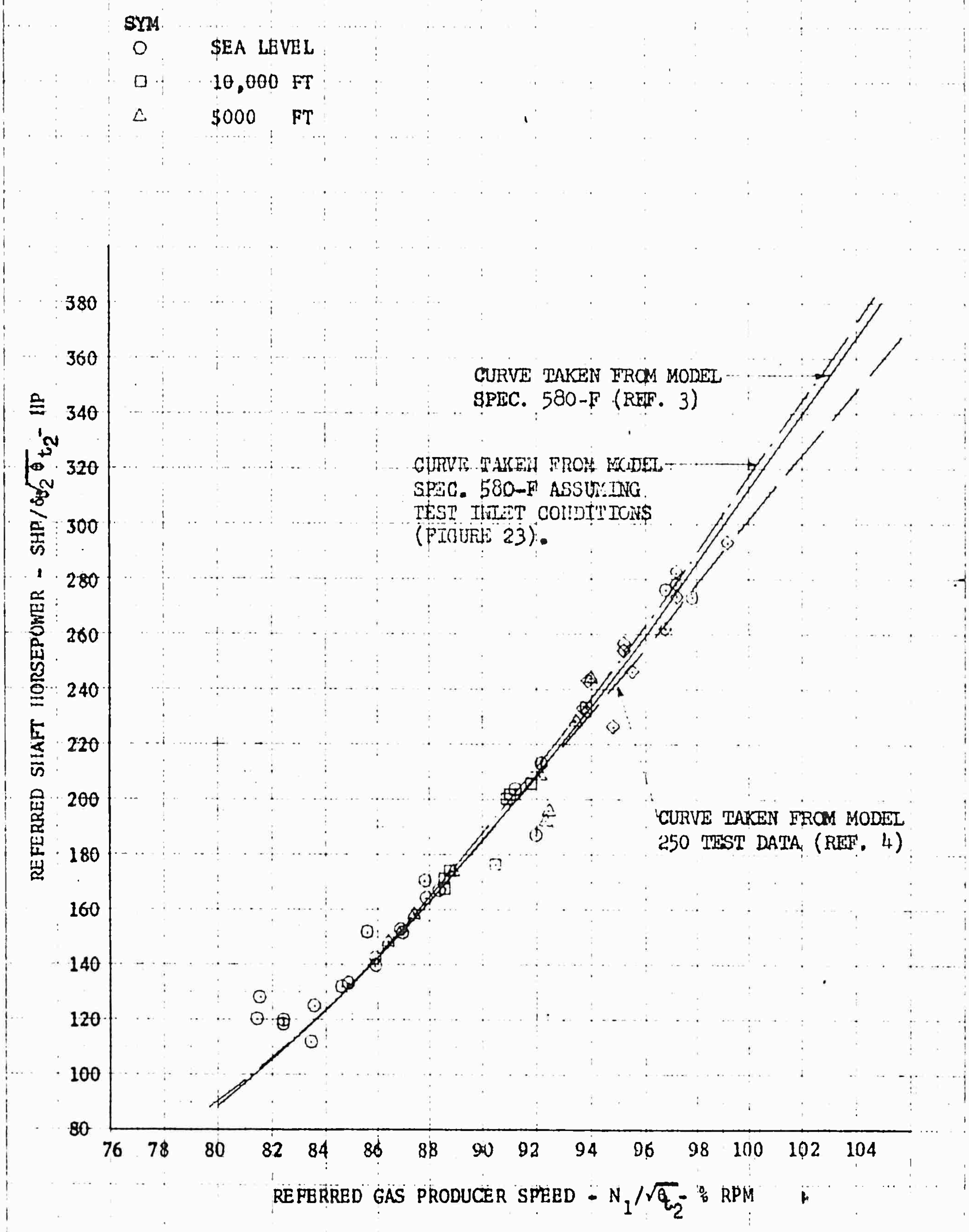
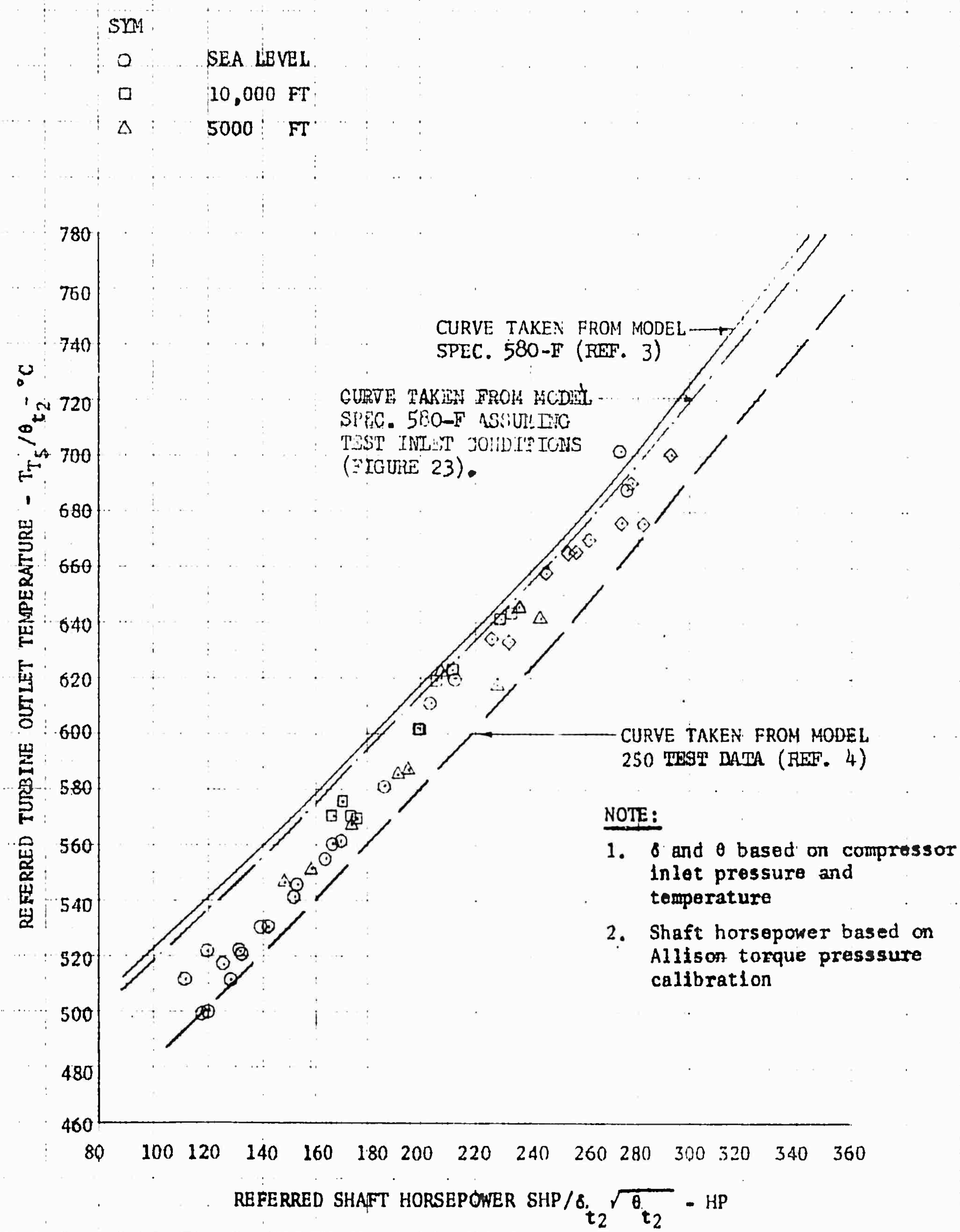


FIGURE NO. 26
ENGINE CHARACTERISTICS
OH-6A S/N 65-12927
T63-A-5A



NOTE:

1. δ and θ based on compressor inlet pressure and temperature
2. Shaft horsepower based on Allison torque presssure calibration

FIGURE NO. 27
ENGINE CHARACTERISTICS
OH-6A S/N 65-12927
T63-A-5A

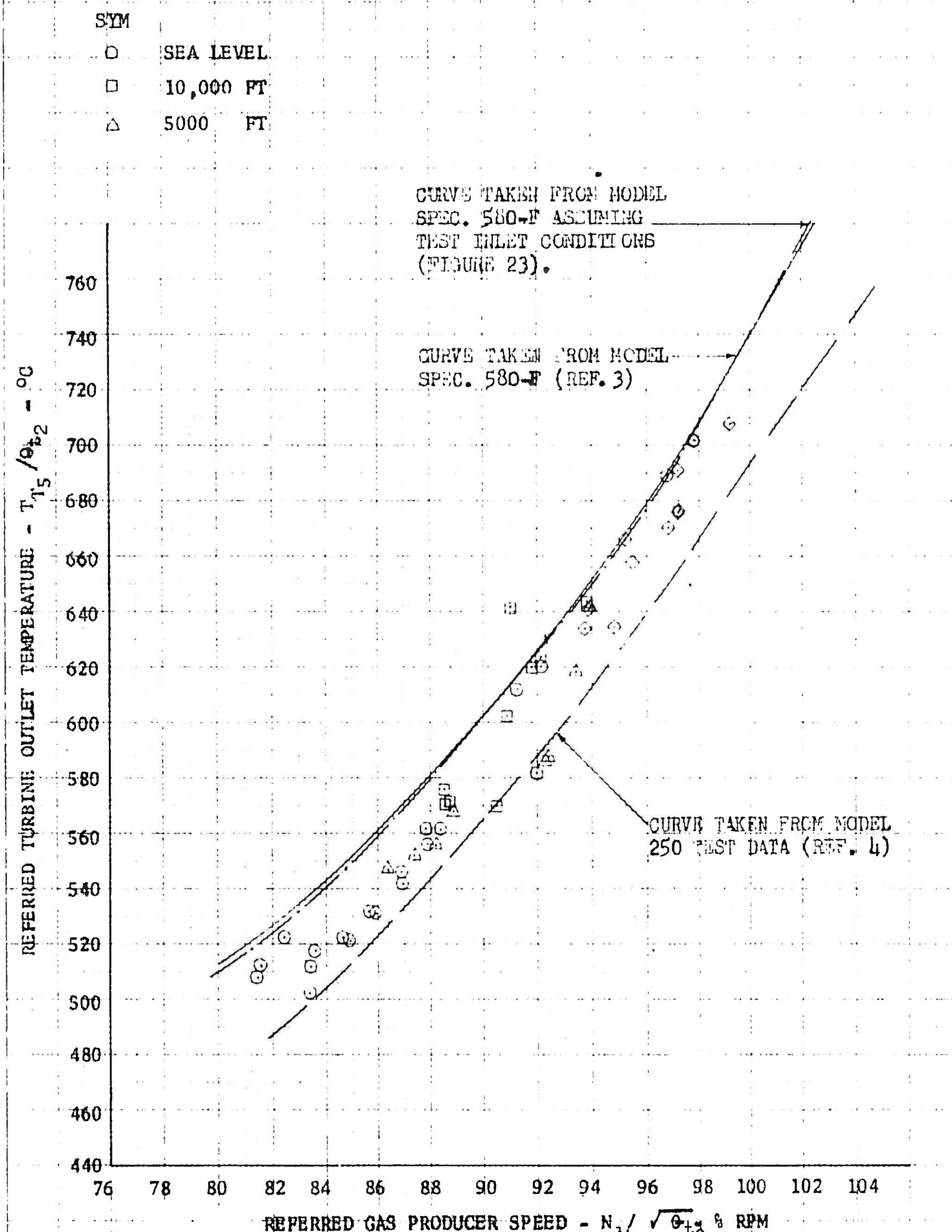


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FIGURE NO. 28
CYCLIC CONTROL PATTERN
CH-6A S/N 65-12927

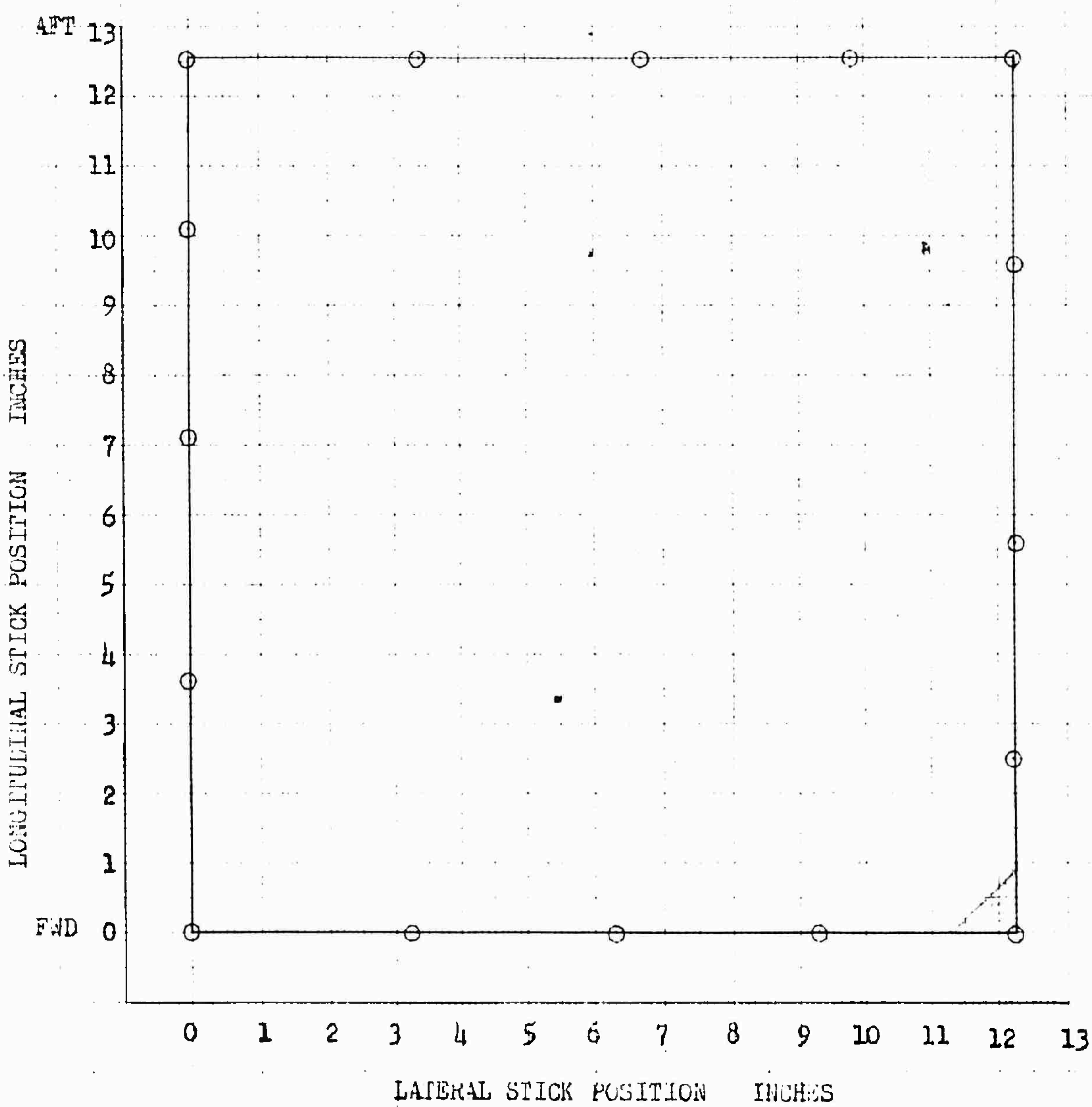

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FIGURE NO. 29
CONTROL POSITIONS IN FORWARD FLIGHT
OH-6A S/N 65-12927
CLEAN CONFIGURATION

SYM	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
○	5550	2290	483	104.0 (aft)	1.4 (rt)	Level Flight
□	5060	2300	483	100.0 (mid)	0 (mid)	Level Flight
△	5270	2290	483	97.0 (fwd)	0 (mid)	Level Flight

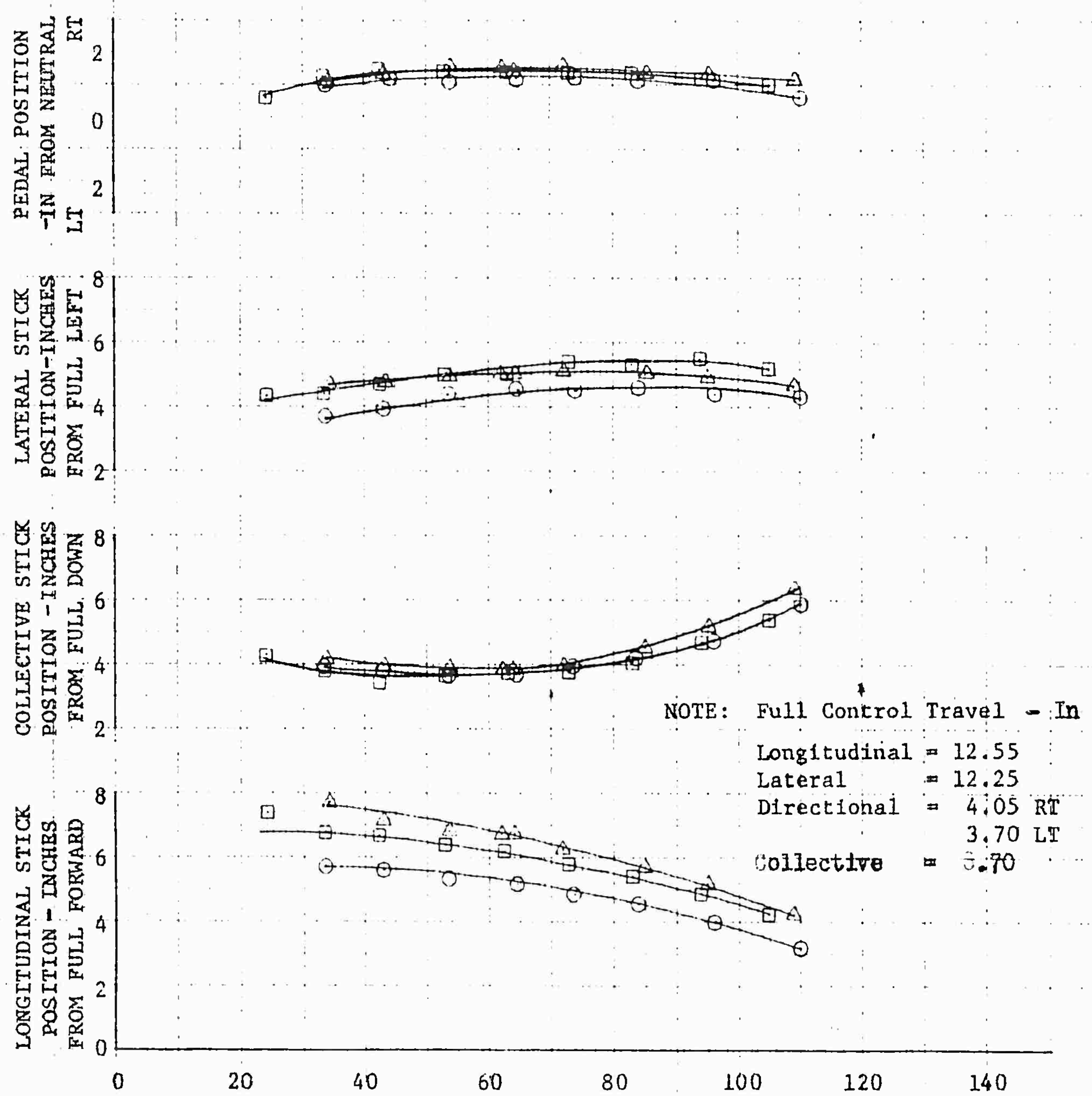
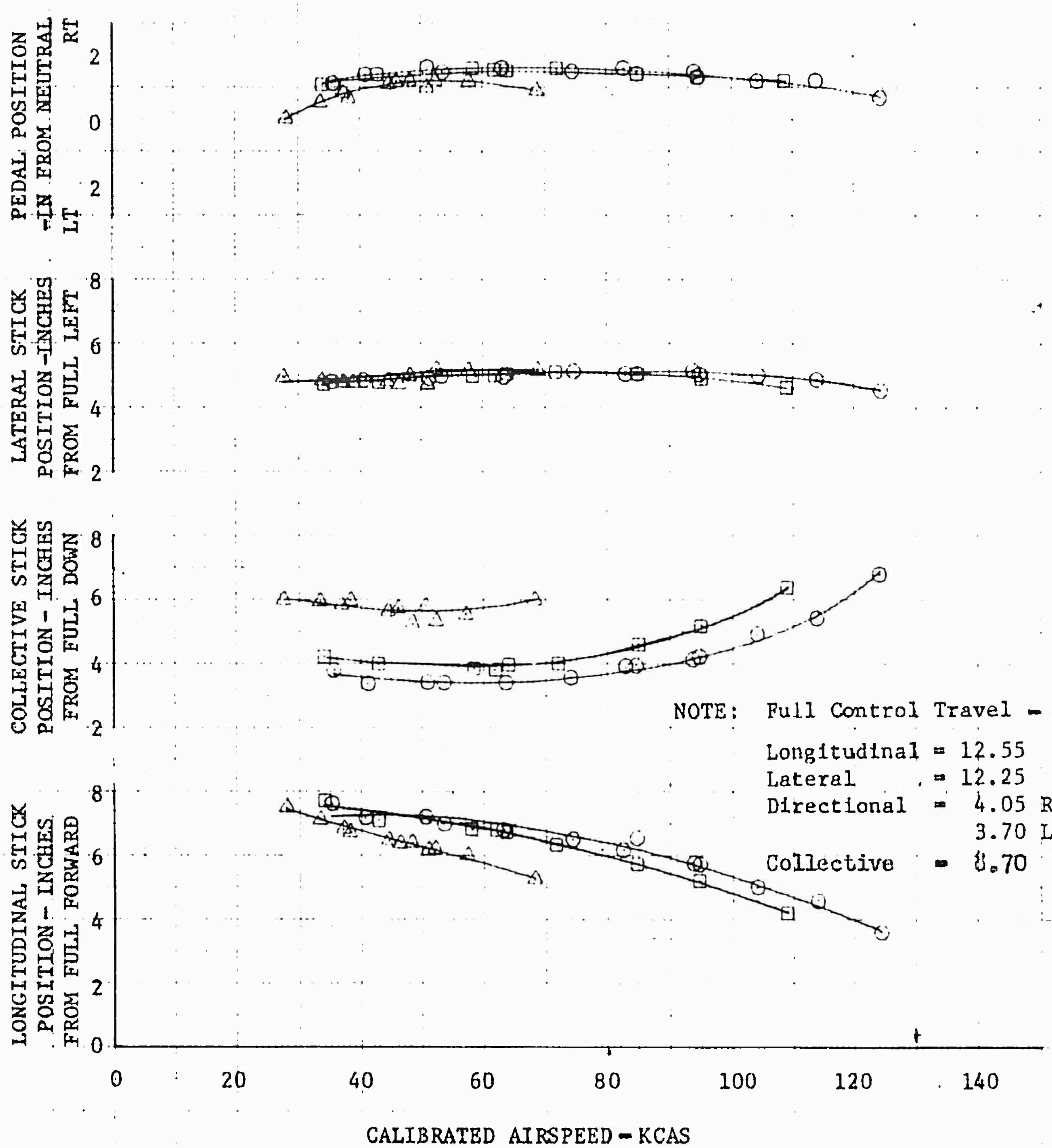


FIGURE NO. 30
CONTROL POSITIONS IN FORWARD FLIGHT
OH-6A S/N 65-12927
CLEAN CONFIGURATION

SYM	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
○	1000	2260	483	97.0 (fwd)	0 (mid)	Level Flight
□	5270	2290	483	97.0 (fwd)	0 (mid)	Level Flight
△	13800	2300	483	97.0 (fwd)	0 (mid)	Level Flight



NOTE: Full Control Travel - In

Longitudinal = 12.55
Lateral = 12.25
Directional = 4.05 RT
3.70 LT
Collective = 8.70

FIGURE NO. 31
CONTROL POSITIONS IN FORWARD FLIGHT
OH-6A S/N 65-12927

SYM	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	CONF.	FLIGHT CONDITION
□	5380	2500	483	97.0 (fwd)	1.4 (lt)	Armed	Level Flight
○	5270	2290	483	97.0 (fwd)	0 (mid)	Clean	Level Flight

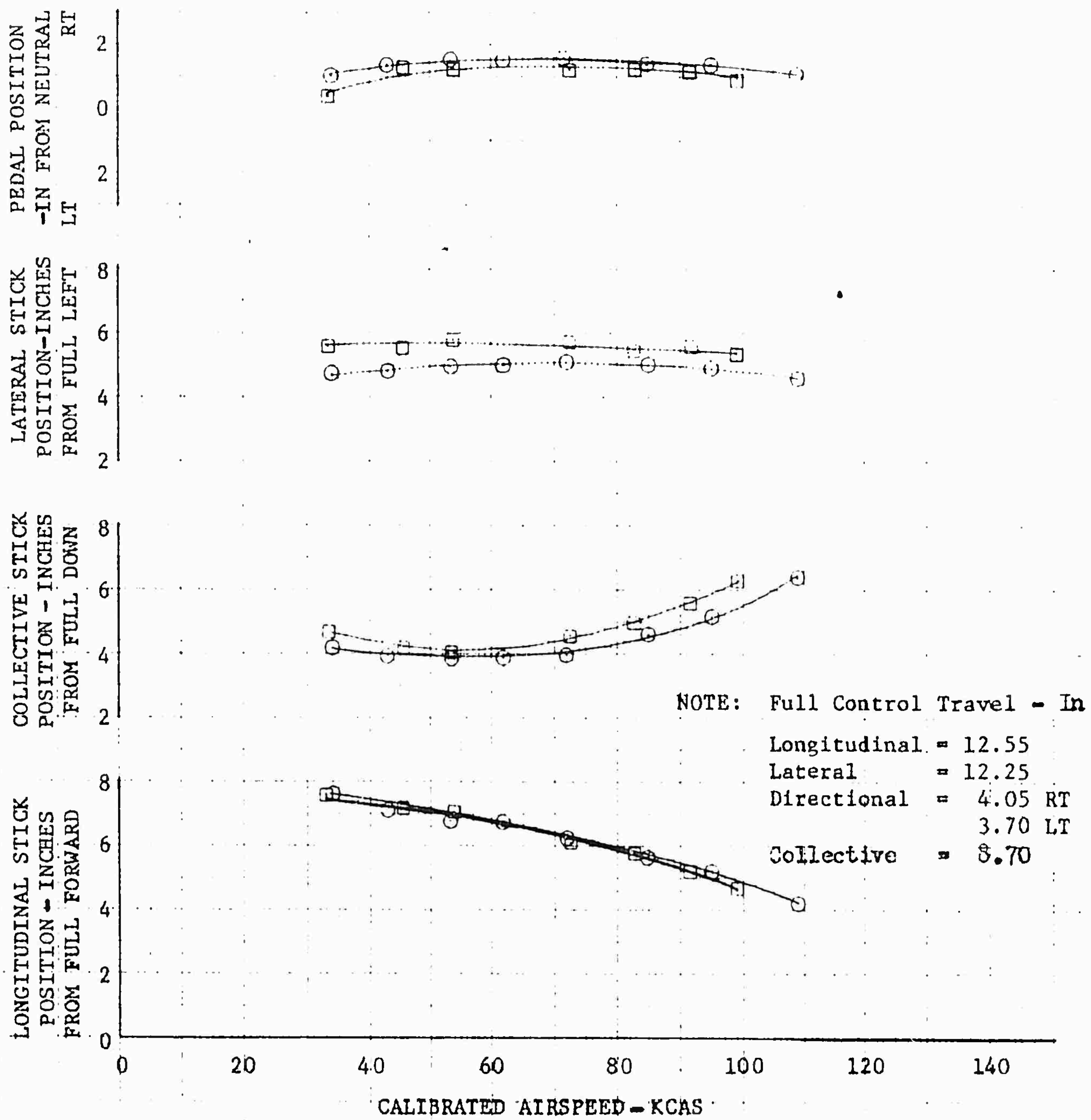


FIGURE NO. 32

STATIC LONGITUDINAL COLLECTIVE FIXED STABILITY

OH-6A S/N 95-12927

CLEAN CONFIGURATION

SYM	TRIM AIRSPEED KCAS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION LONG	C.G. LOCATION LAT	FLIGHT CONDITION
○	35.0	6420	2270	483	101.3(mid)	0(mid)	Level Flight
□	81.0	6000	2280	483	101.3(mid)	0(mid)	Level Flight
△	97.0	6660	2250	483	101.3(mid)	0(mid)	Level Flight

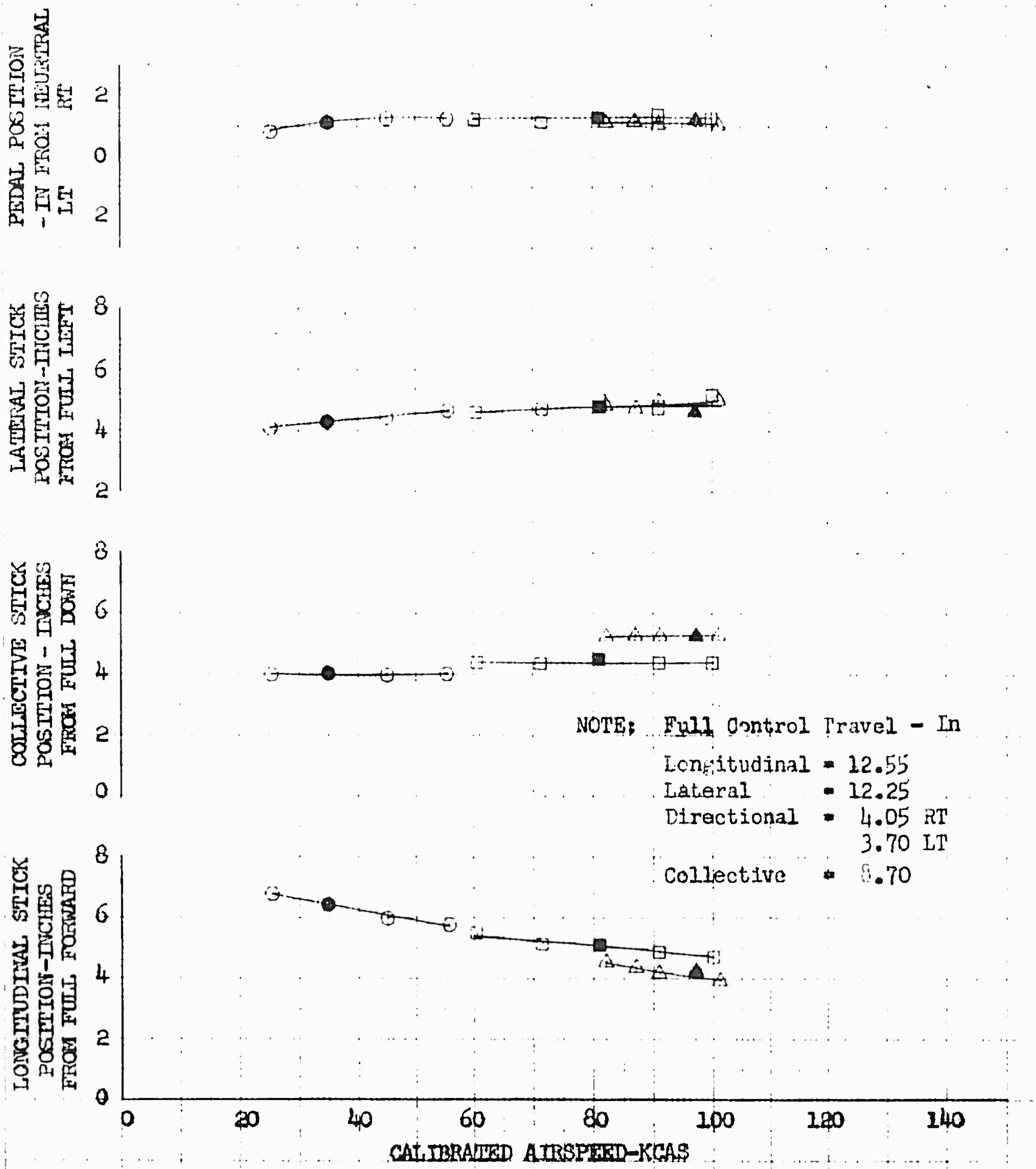
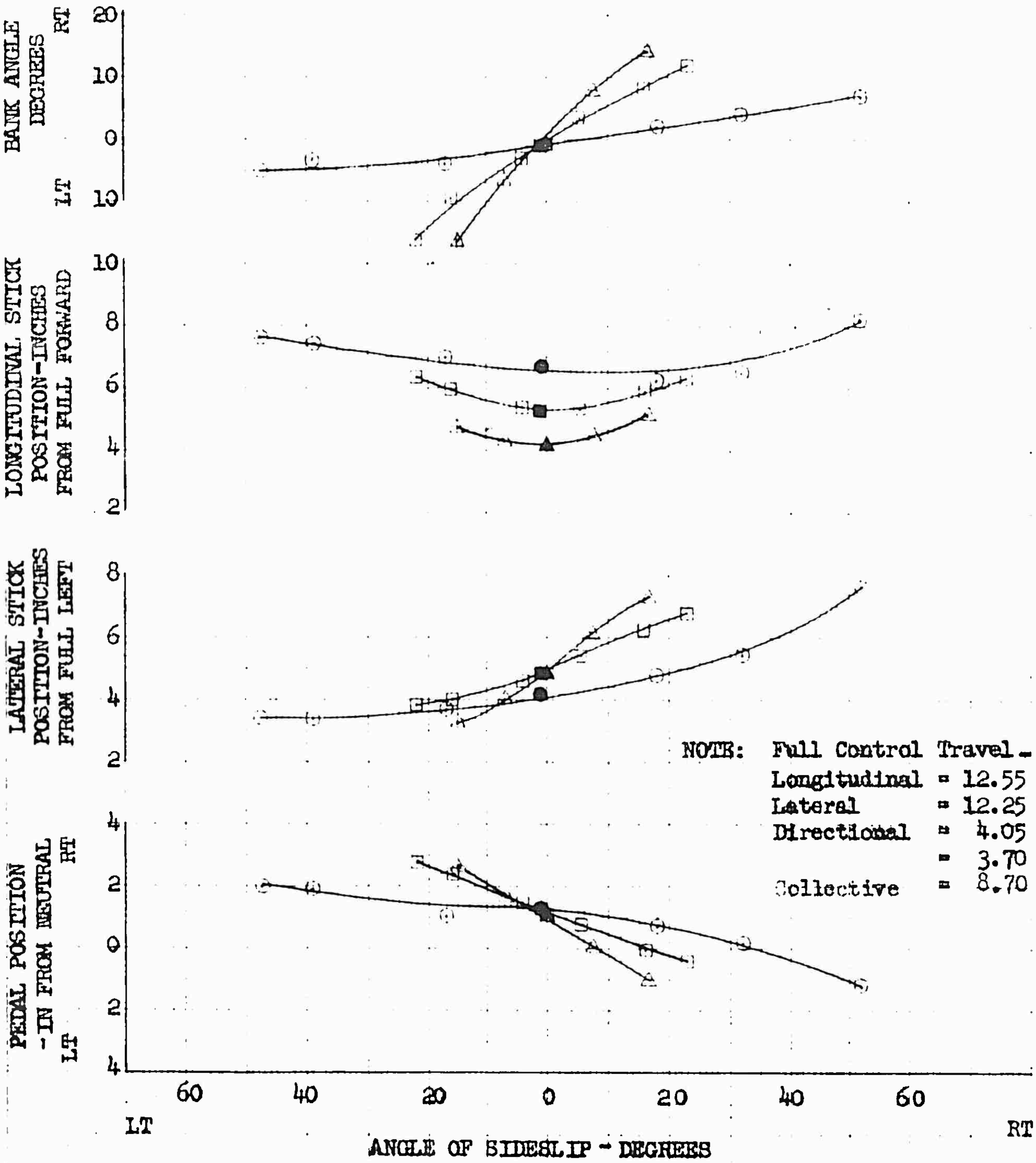


FIGURE NO. 33
STATIC LATERAL-DIRECTIONAL STABILITY
OH-6A S/N 65-12927
CLEAN CONFIGURATION

SYM	TRIM AIRSPEED KCAS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
○	35.5	5510	2280	483	100.6(mid)	0(mid)	Level Flight
□	81.0	5750	2310	483	100.6(mid)	0(mid)	Level Flight
△	102.0	5870	2300	483	100.6(mid)	0(mid)	Level Flight

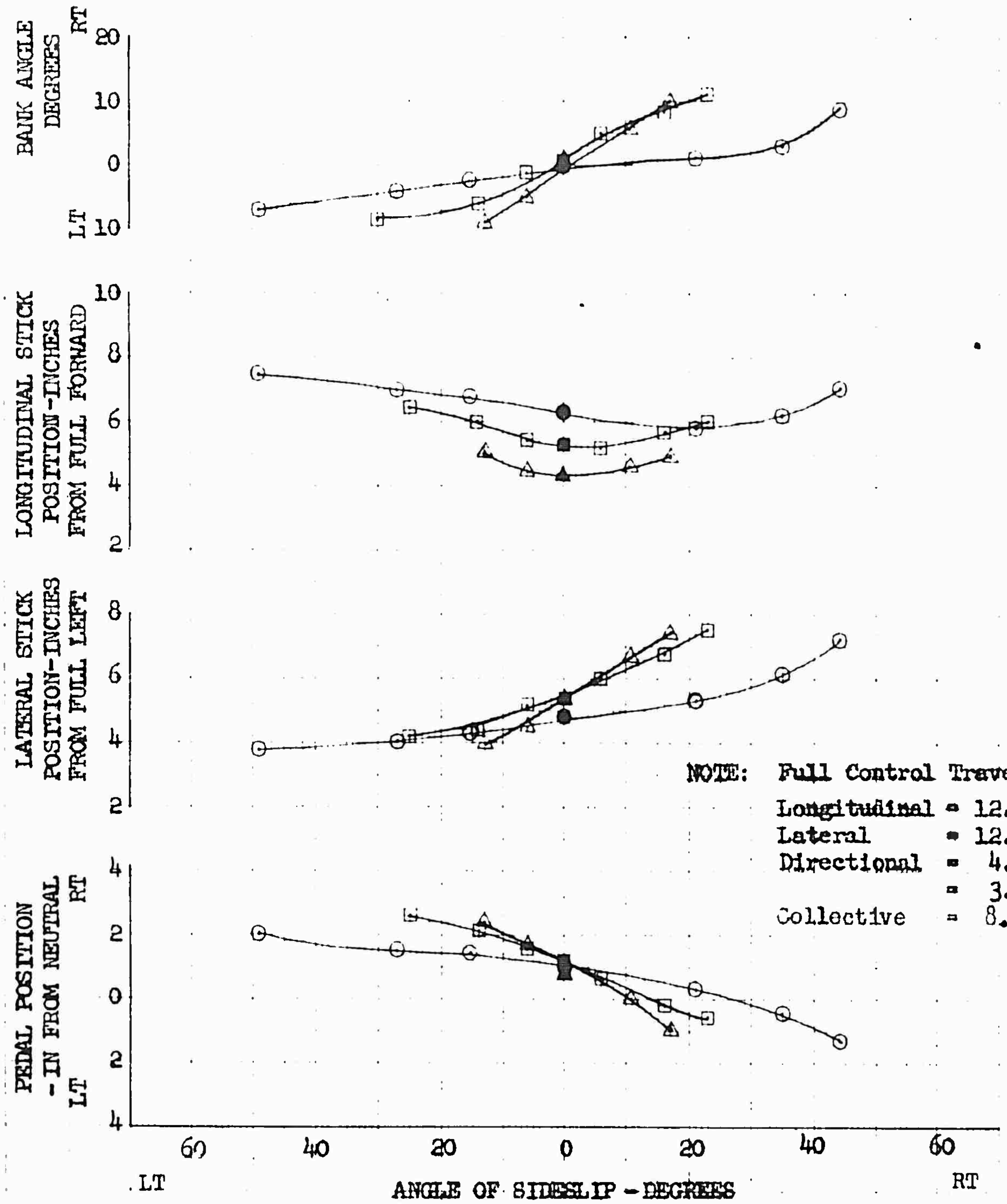


NOTE: Full Control Travel - In
Longitudinal = 12.55
Lateral = 12.25
Directional = 4.05 RT
= 3.70 LT
Collective = 8.70

FIGURE NO. 34
STATIC LATERAL-DIRECTIONAL STABILITY

OH-6A S/N 65-12927
XM-27-E1 IN STOWED POSITION

SYM	TRIM AIRSPEED KCAS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
○	35.5	6000	2460	483	101.0(mid)	1.5(LT)	Level Flight
□	71.5	5890	2490	483	101.0(mid)	1.5(LT)	Level Flight
△	90.0	5960	2480	483	101.0(mid)	1.5(LT)	Level Flight



NOTE: Full Control Travel - In
Longitudinal = 12.55
Lateral = 12.25
Directional = 4.05 RT
= 3.70 LT
Collective = 8.70

FIGURE NO. 35
CONTROL POSITIONS DURING SIDEWARD FLIGHT
OH-6A S/N 65-12967
XM-27-E1 IN STOWED POSITION IGE (10 FEET)

SYM	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG.	C.G. LOCATION IN LAT.
○	1070	2430	483	97.1 (FWD)	2.0 (LT)

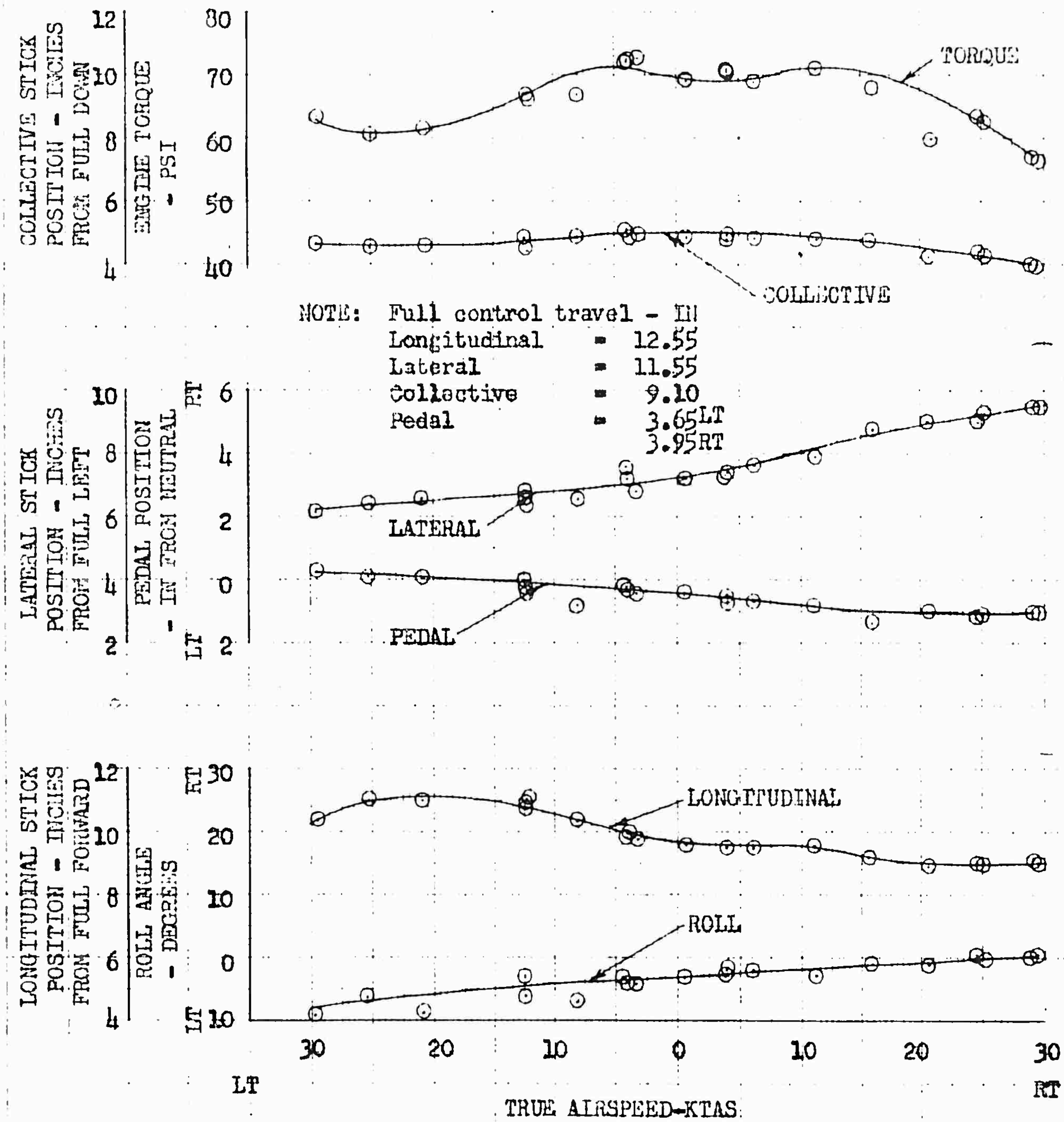
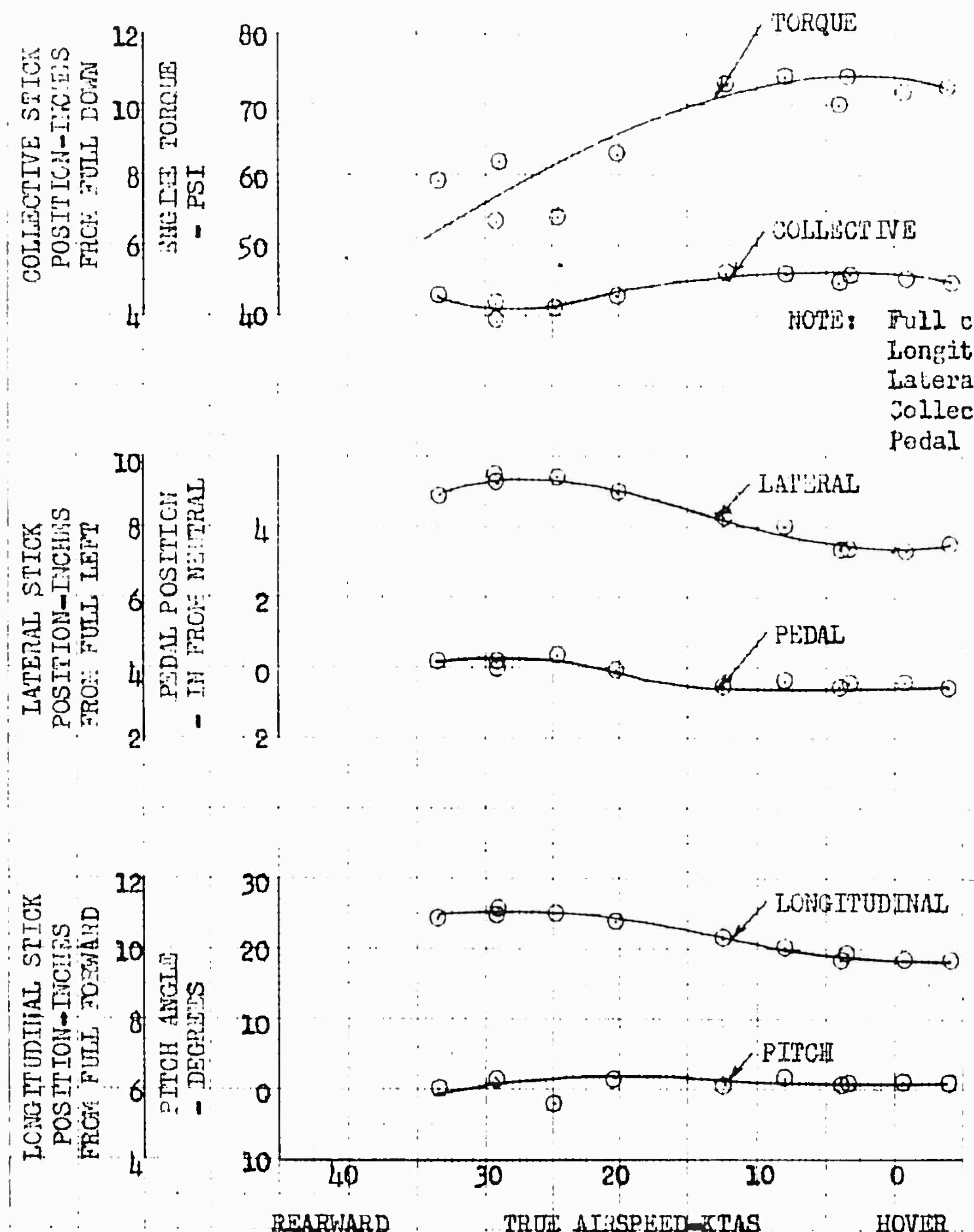


FIGURE NO. 36
CONTROL POSITIONS DURING REARWARD FLIGHT
OH-6A S/N 65-12967

XM-27-E1 IN STOWED POSITION IGE (10 FEET)

SYM	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG.	C.G. LOCATION IN LAT.
○	1260	2400	483	97.1(FWD)	2.0(LT)



NOTE: Full control travel - IN
Longitudinal = 12.55
Lateral = 11.55
Collective = 9.10
Pedal = 3.65 LT
3.95 RT

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FIGURE
AFT LONGITUDI
OH-6A S/N
CLEAN CONFIG

LEVEL FLIGHT
TRIM AIRSPEED = 81.0 KCAS
DENSITY ALTITUDE = 5000 FT

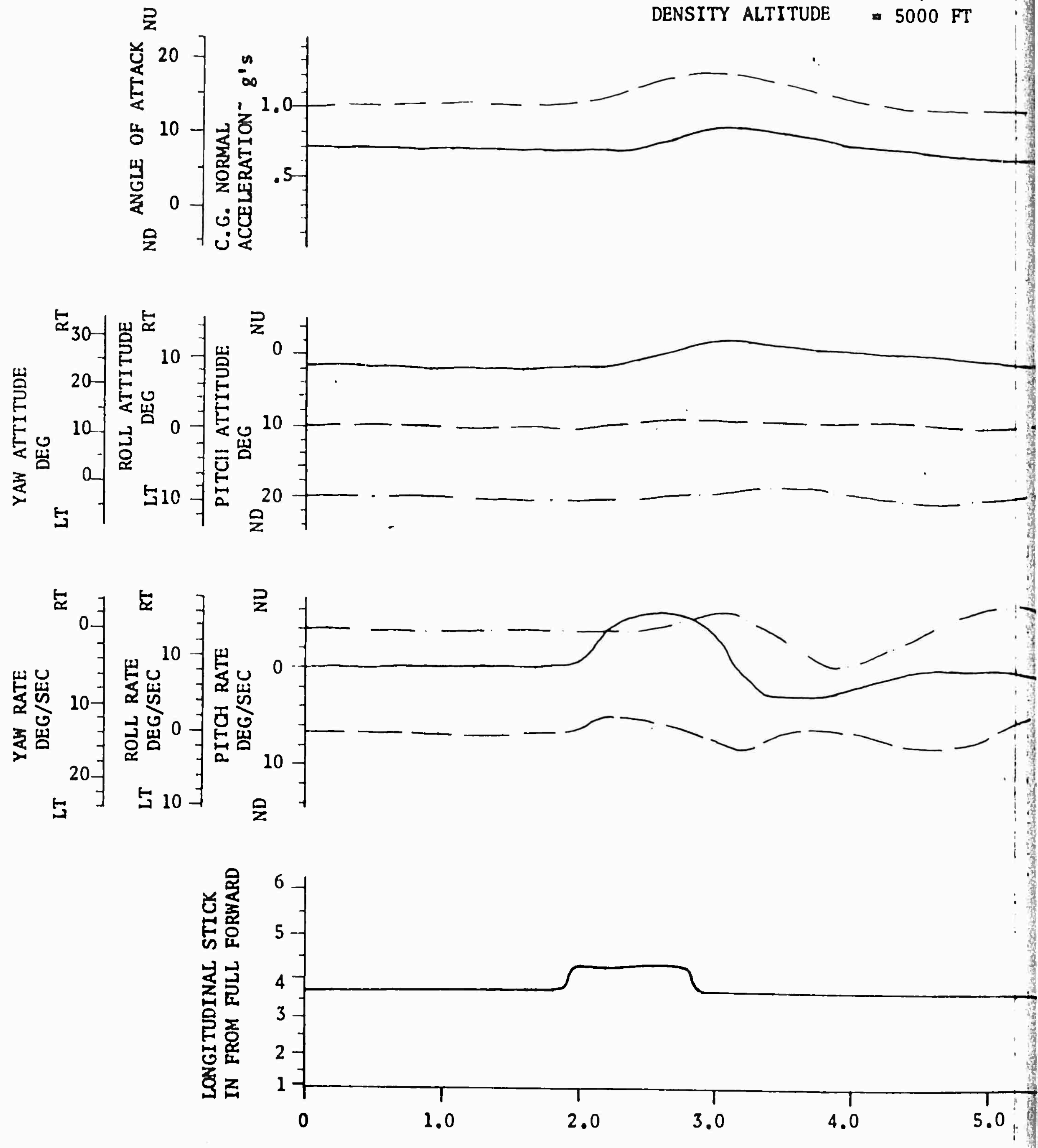

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FIGURE NO. 37

T LONGITUDINAL PULSE

OH-6A S/N 65-12927

CLEAN CONFIGURATION

81.0 KCAS	GROSS WEIGHT	= 2290 LBS
5000 FT	ROTOR SPEED	= 483 RPM
	C.G. LOCATION	= STATION 100.6 (MID)

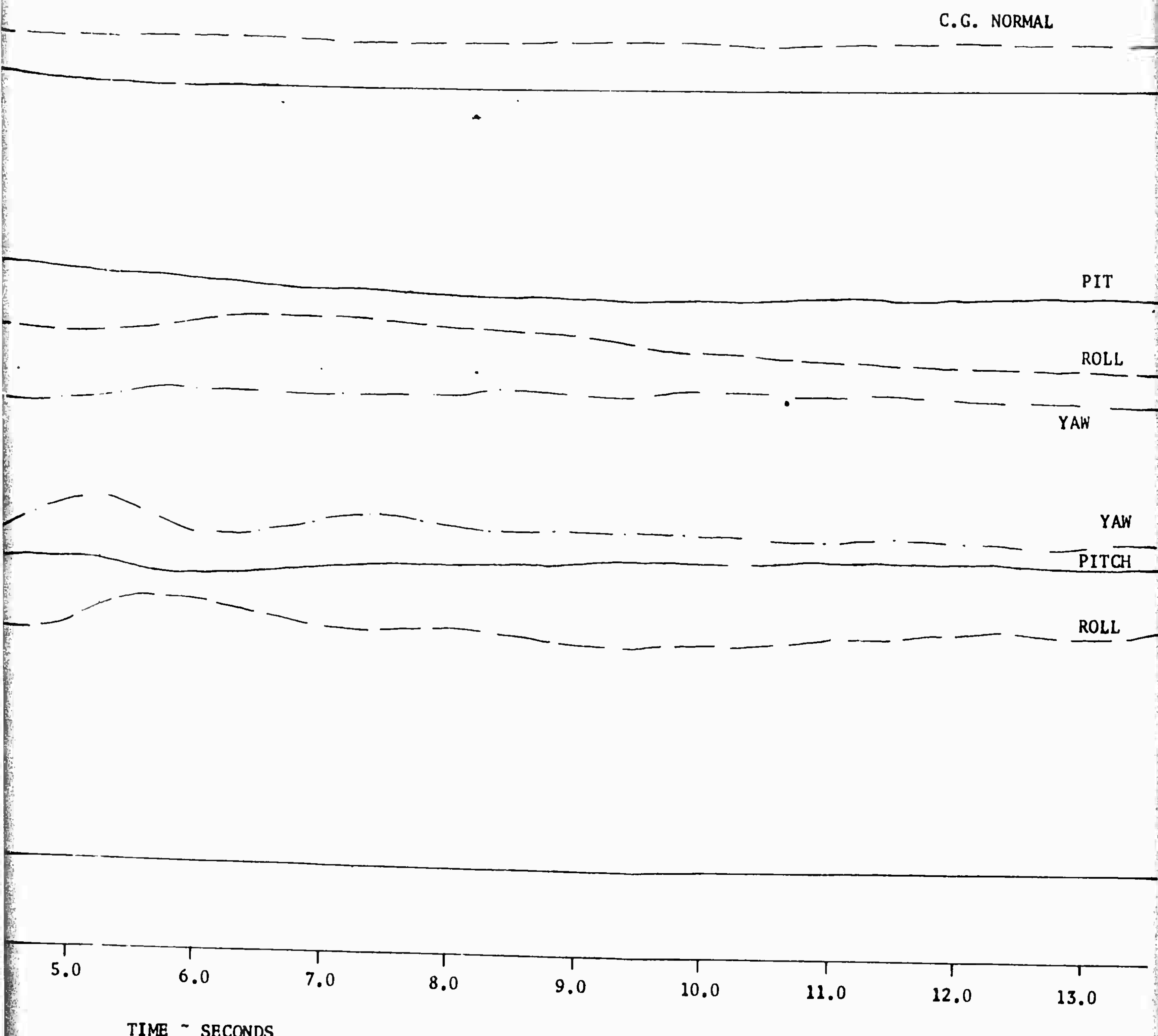

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FIGURE NO.
RIGHT LATERAL P
OH-6A S/N 65-
CLEAN CONFIGURA

LEVEL FLIGHT
TRIM AIRSPEED = 81.0 KCAS
DENSITY ALTITUDE = 5000 FT

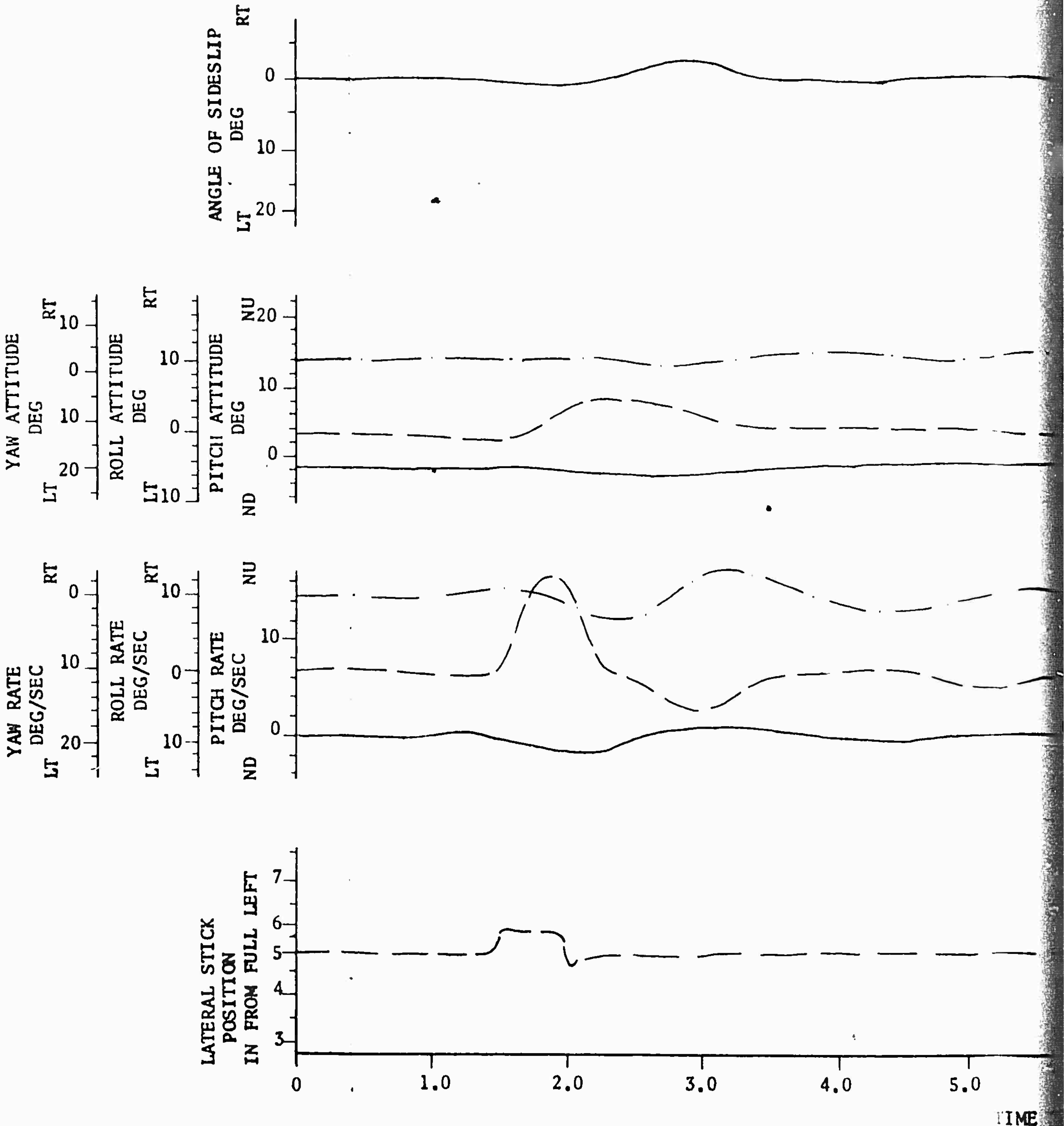

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FIGURE NO. 38
RIGHT LATERAL PULSE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

= 81.0 KCAS
= 5000 FT

GROSS WEIGHT = 2290 LBS
ROTOR SPEED = 483 RPM
C.G. LOCATION = STATION 100.6 (MID)

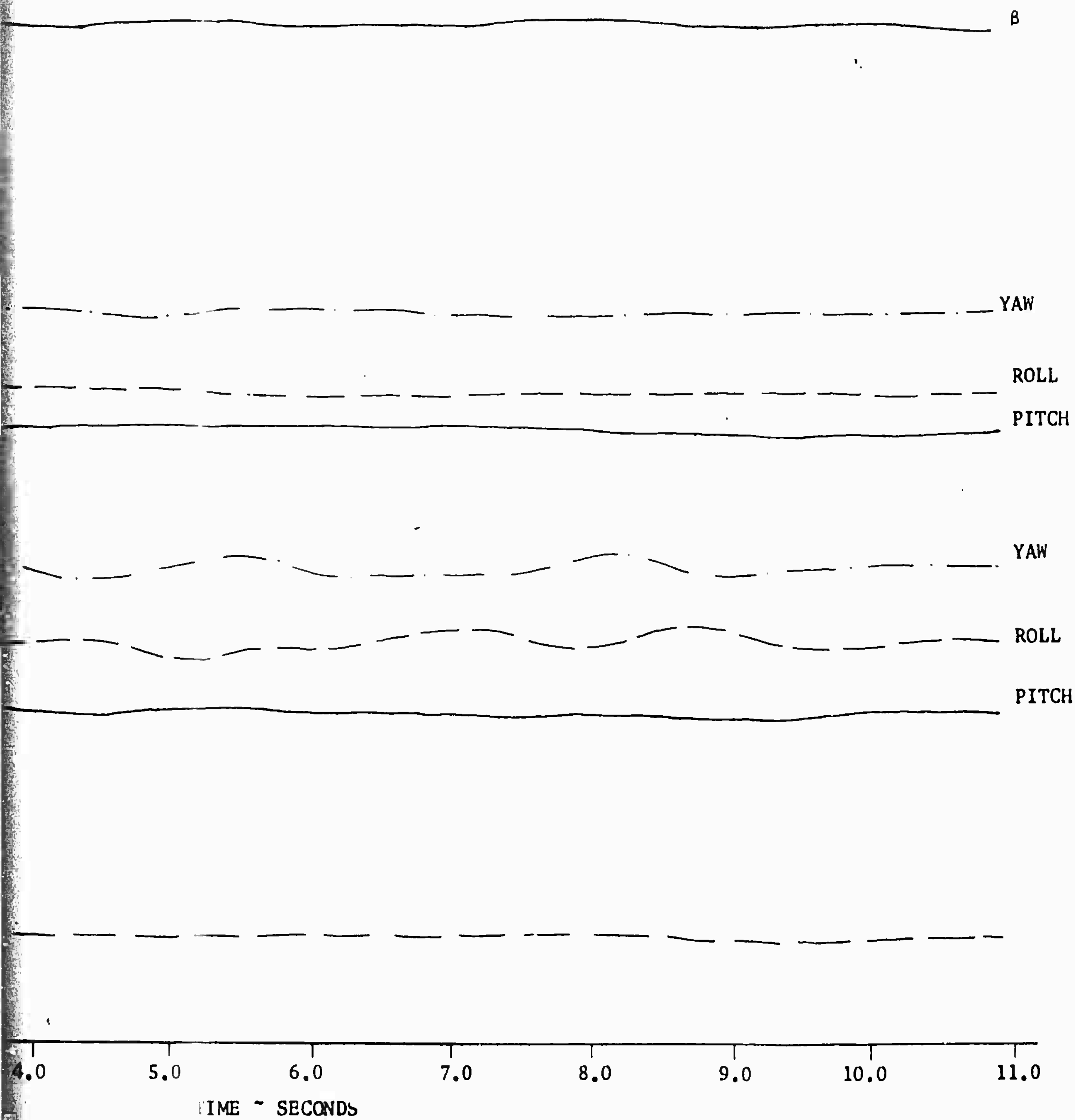

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FIGURE NO.
RIGHT DIRECTIONAL
OH-6A S/N 65-
CLEAN CONFIGURA

LEVEL FLIGHT
TRIM AIRSPEED = 81.0 KCAS
DENSITY ALTITUDE = 5000 FT

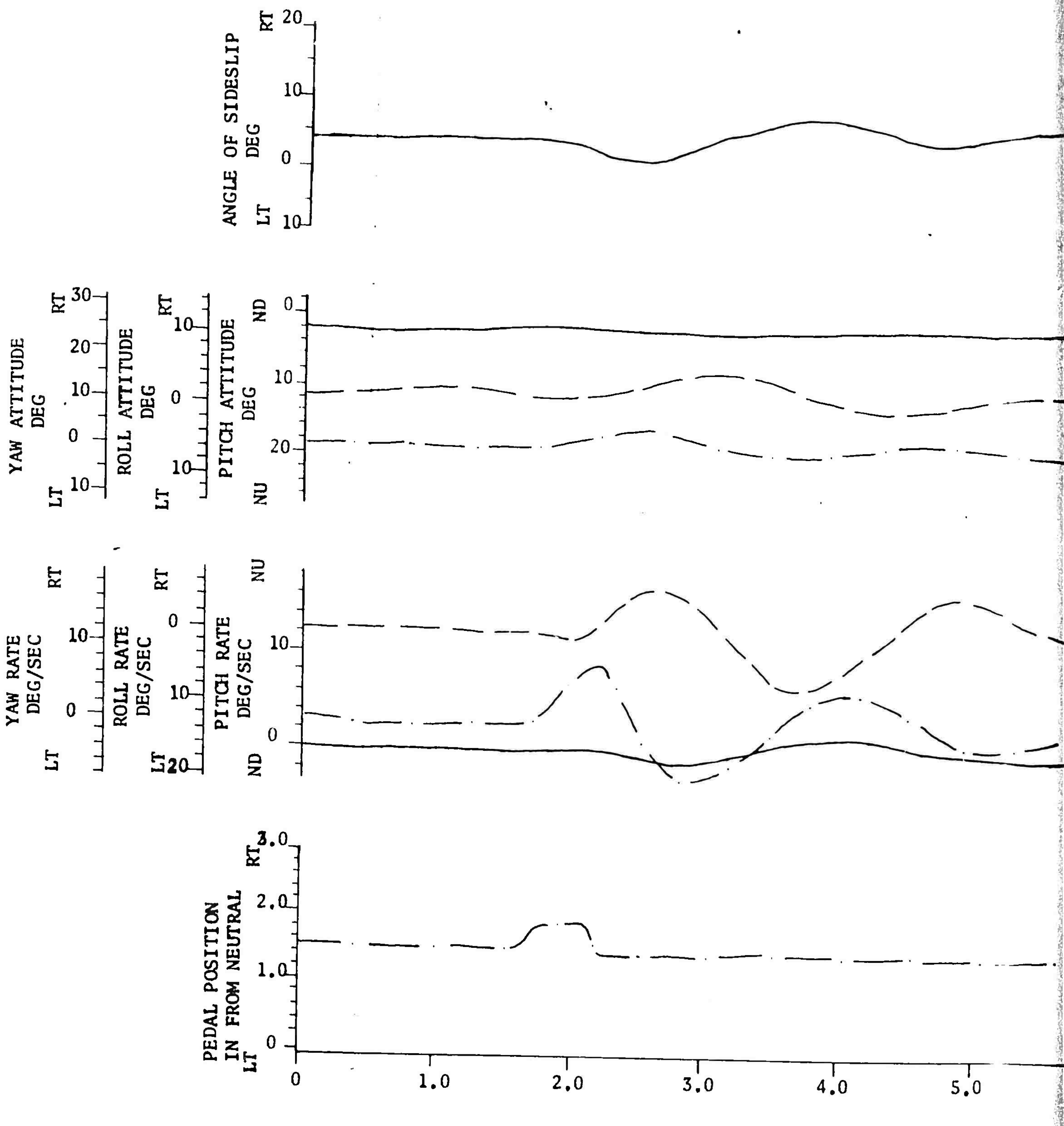

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FIGURE NO. 39
RIGHT DIRECTIONAL PULSE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

81.0 KCAS	GROSS WEIGHT	= 2290 LBS
5000 FT	ROTOR SPEED	= 483 RPM
	C.G. LOCATION	= STATION 100.6 (MID)

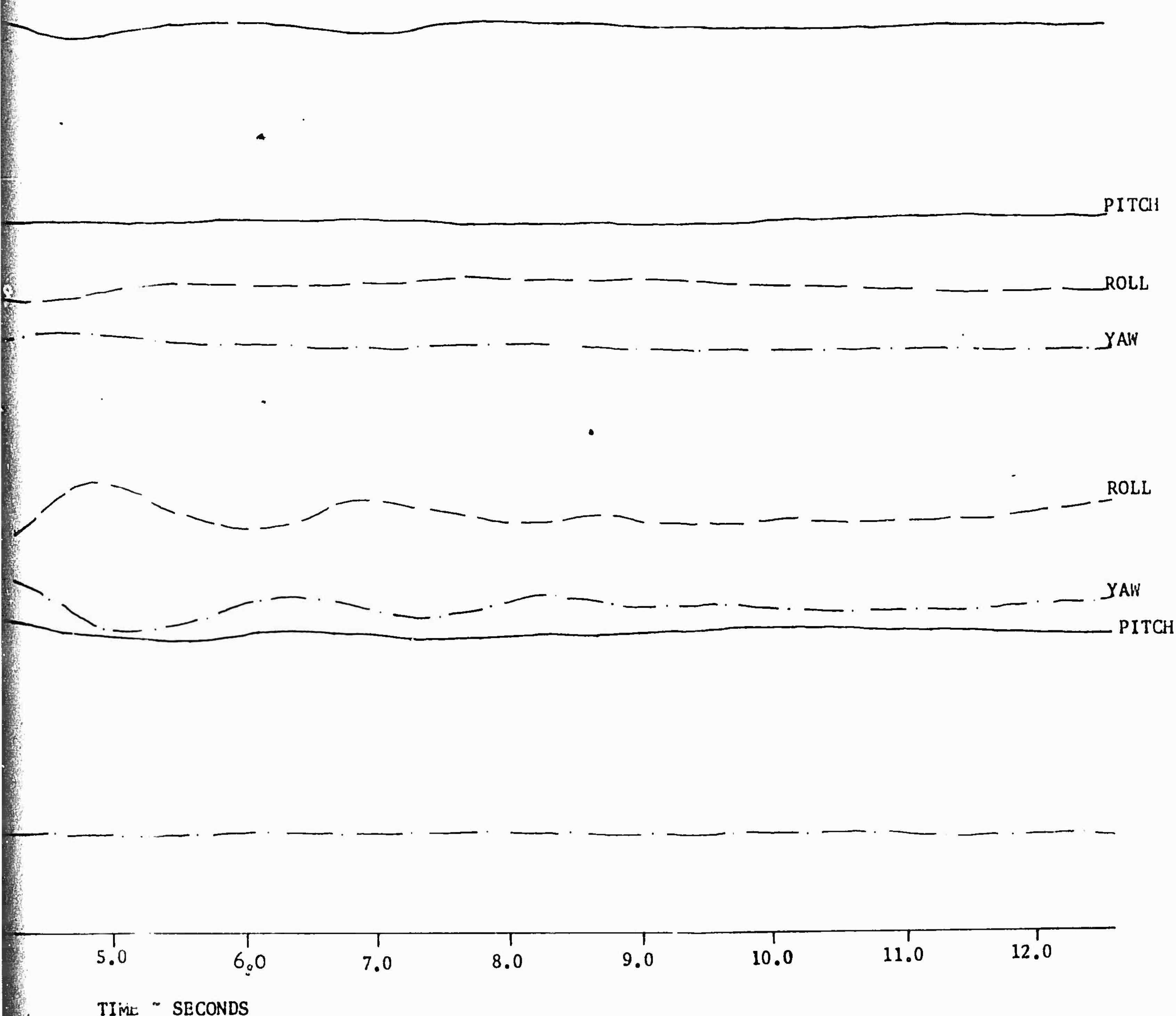

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FIGURE NO. 40
SUMMARY OF CONTROL RESPONSE
OH-6A S/N 65-12927

SYM	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	CONF.	FLIGHT CONDITION
○	5000	2290	483	100.6(mid)	0(mid)	Clean	Level Flight
□	6000	2300	483	97.2(fwd)	0(mid)	Clean	Level Flight
△	6000	2500	483	101.3(mid)	1.5(lt)	Armed	Level Flight

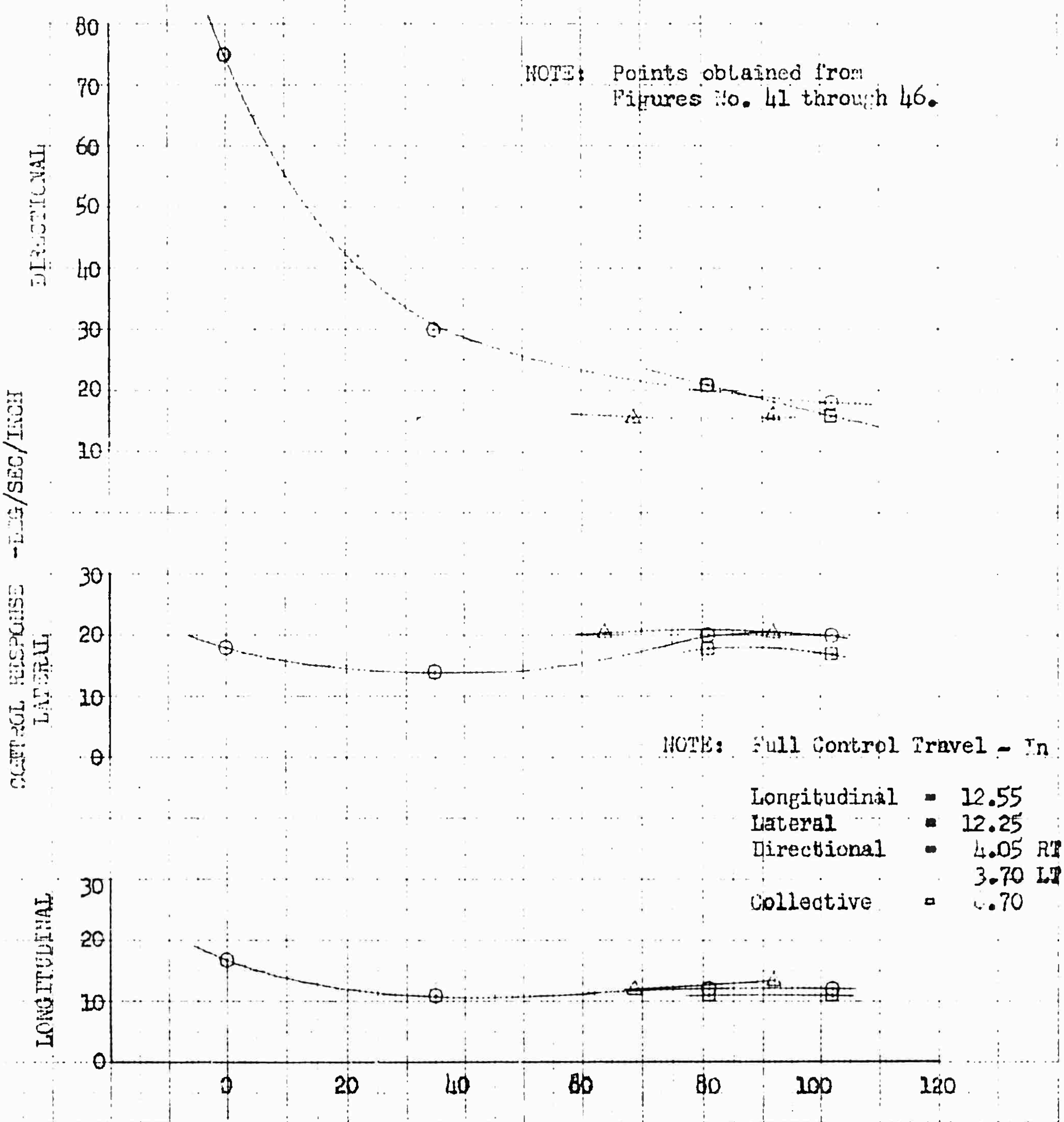


FIGURE NO. 41
LONGITUDINAL CONTROL RESPONSE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

SYM	CALIB. AIRSPEED KTS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	LAT	FLIGHT CONDITION
○	0	3300	2200	483	100.6(mid)	0(mid)	Hover
□	35.5	5000	2270	483	100.6(mid)	0(mid)	Level Flight
△	81.0	5000	2290	483	100.6(mid)	0(mid)	Level Flight
◇	102.0	5400	2250	483	100.6(mid)	0(mid)	Level Flight

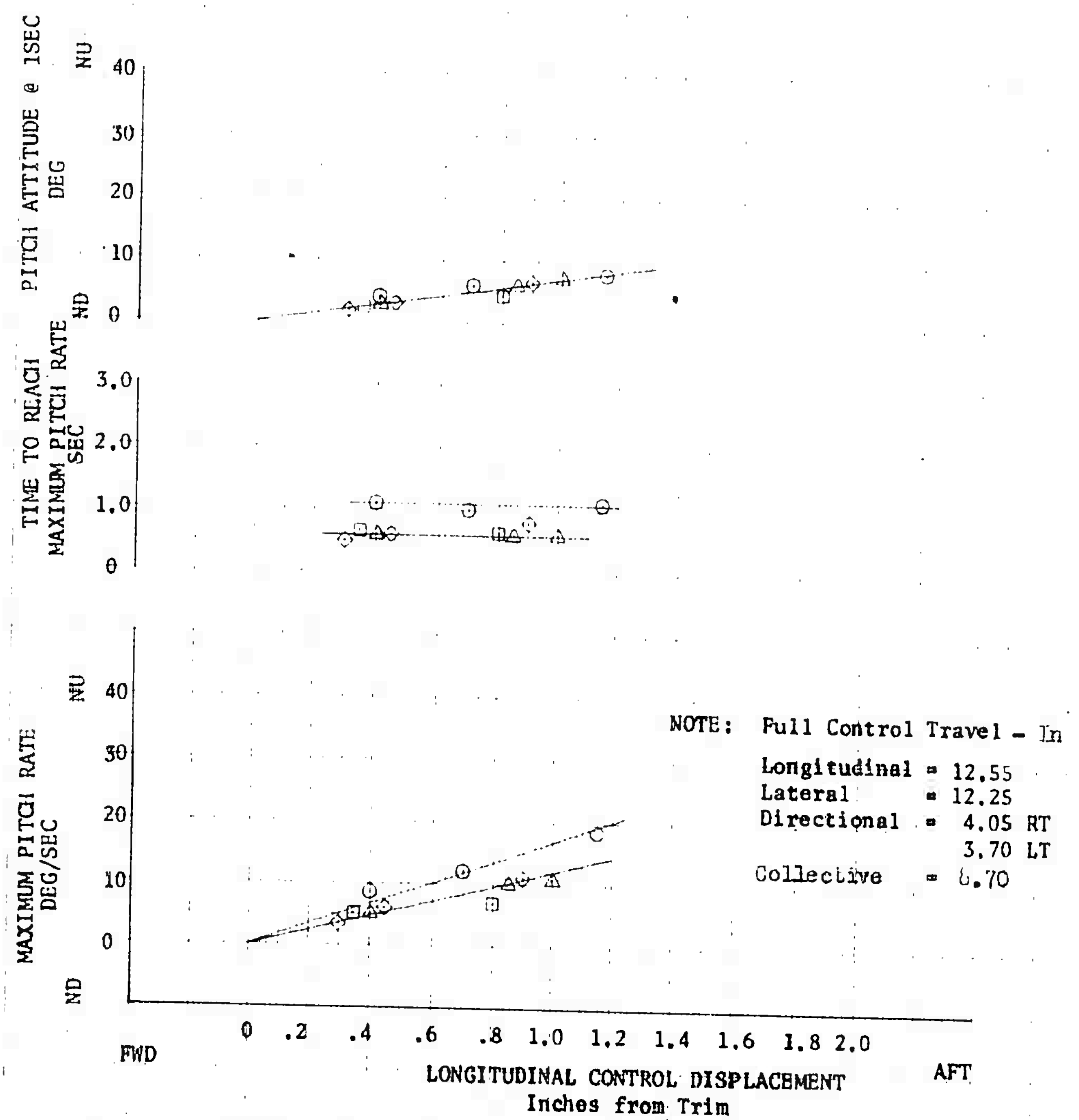


FIGURE NO. 42
LONGITUDINAL CONTROL RESPONSE
OH-6A S/N 65-12927
ARMED-CLEAN COMPARISON

SYM	CALIB AIRSPEED KTS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	CONF.	FLIGHT CONDITION
△	81	6000	2300	483	97.2(fwd)	0(mid)	Clean	Level Flight
◇	102	6000	2300	483	97.2(fwd)	0(mid)	Clean	Level Flight
○	74	6000	2500	483	101.3(mid)	1.5(lt)	Armed	Level Flight
□	92	6000	2500	483	101.3(mid)	1.5(lt)	Armed	Level Flight

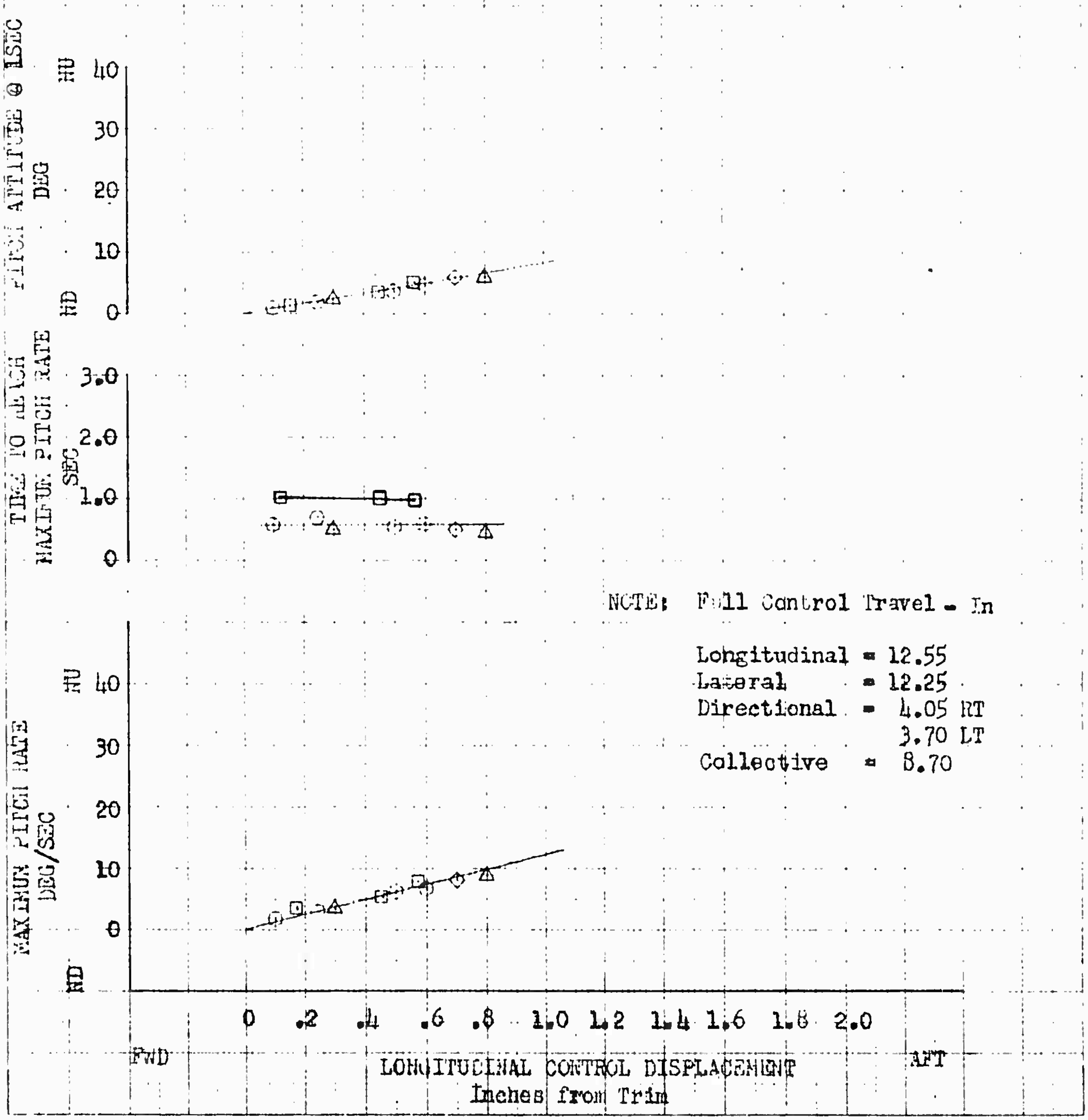
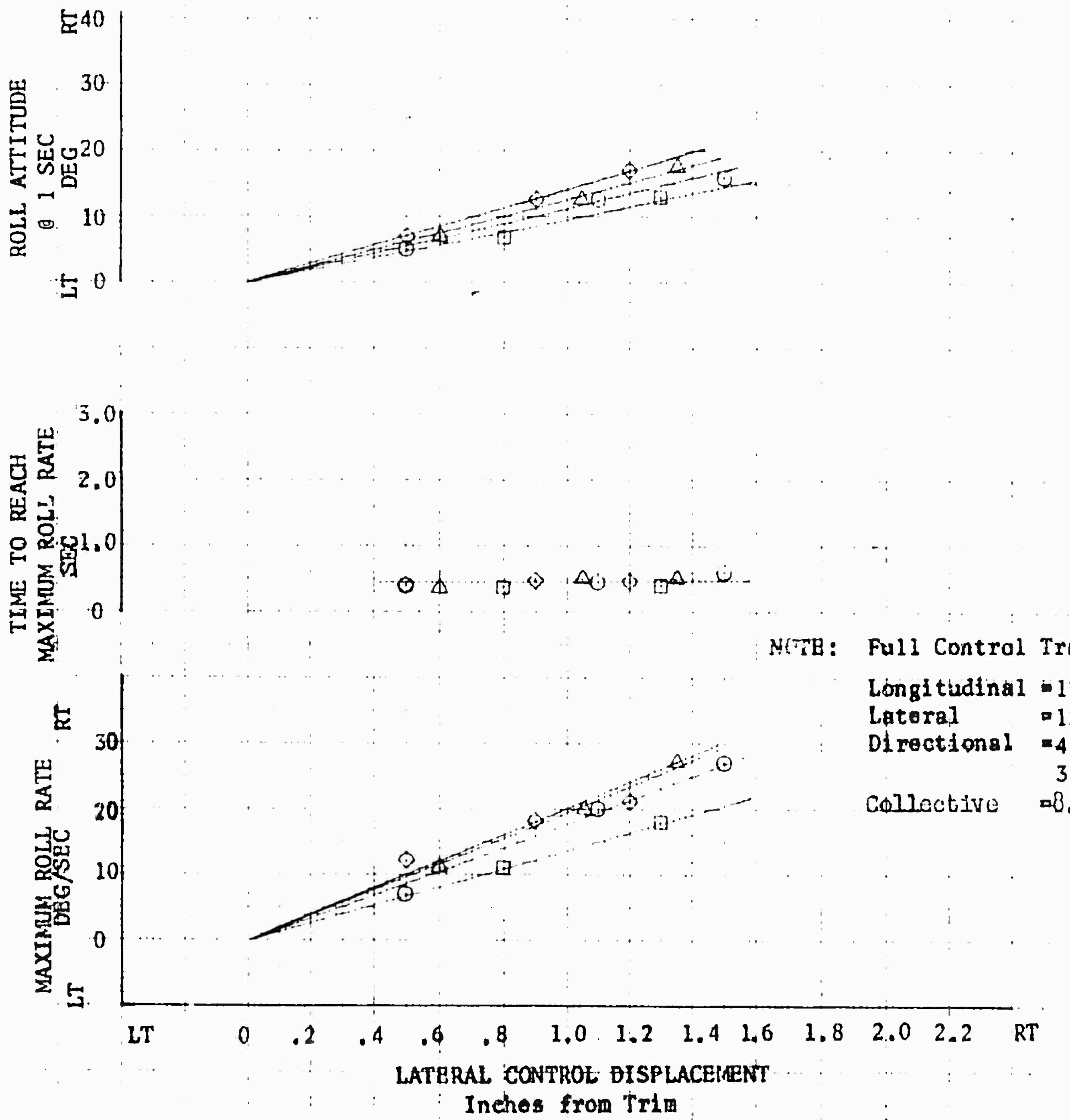


FIGURE NO. 43
LATERAL CONTROL RESPONSE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

SYM	CALIB. AIRSPEED KTS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
○	0	3300	2200	483	100.6(mid)	0(mid)	Hover
□	35.5	5000	2270	483	100.6(mid)	0(mid)	Level Flight
△	81.0	5000	2290	483	100.6(mid)	0(mid)	Level Flight
◇	102.0	5400	2250	483	100.6(mid)	0(mid)	Level Flight



NOTE: Full Control Travel-In

Longitudinal =12.55
Lateral =12.25
Directional =4.05 RT
3.70 LT
Collective =8.70

FIGURE NO. 44
LATERAL CONTROL RESPONSE
OH-6A S/N 65-12927

SYM	CALIB. AIRSPEED KTS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	CONF	FLIGHT CONDITION
△	81	6000	2300	483	97.2(fwd)	0(mid)	Clean	Level Flight
◇	102	6000	2290	483	97.2(fwd)	0(mid)	Clean	Level Flight
○	74	6000	2510	483	101.3(mid)	1.5(lt)	Armed	Level Flight
□	92	6000	2490	483	101.3(mid)	1.5(lt)	Armed	Level Flight

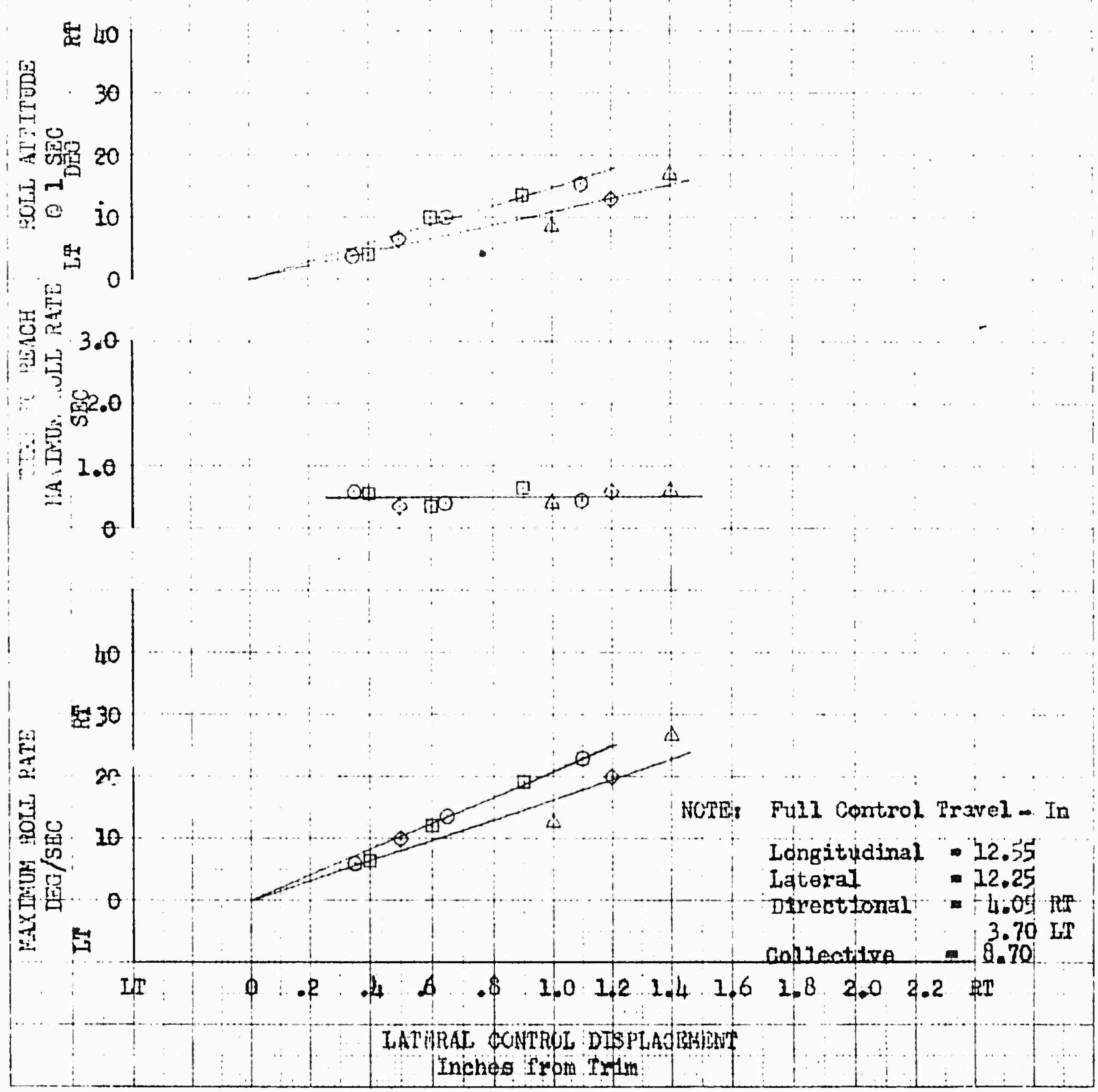
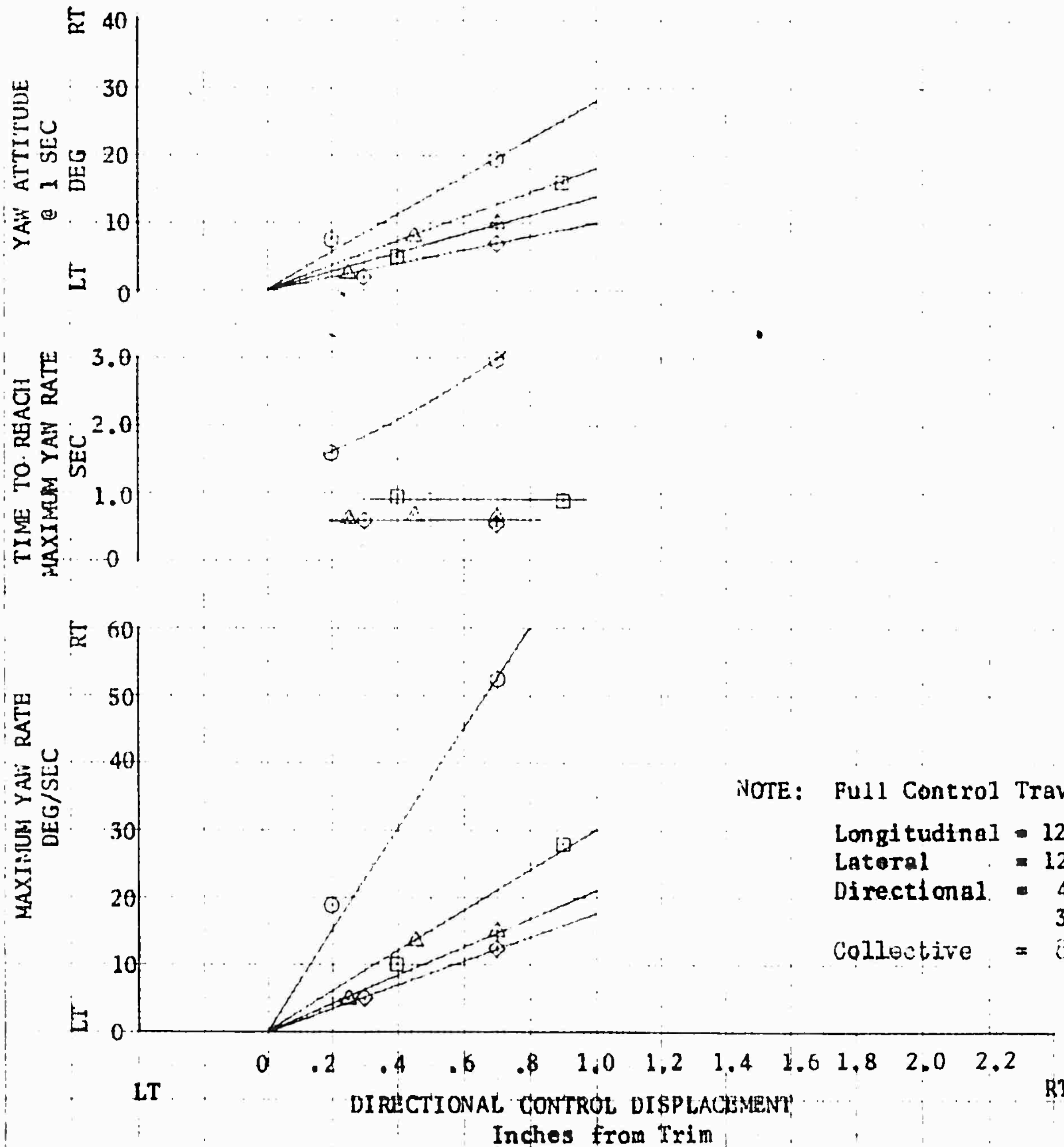


FIGURE NO. 45
DIRECTIONAL CONTROL RESPONSE
OH-6A S/N 65-12927
CLEAN CONFIGURATION

SYM	CALIB. AIRSPEED KTS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G. LOCATION IN LONG	C.G. LOCATION IN LAT	FLIGHT CONDITION
○	0	3300	2200	483	100.6(mid)	0(mid)	Hover
□	35.5	5000	2270	483	100.6(mid)	0(mid)	Level Flight
△	81.0	5000	2290	483	100.6(mid)	0(mid)	Level Flight
◇	102.0	5400	2250	483	100.6(mid)	0(mid)	Level Flight

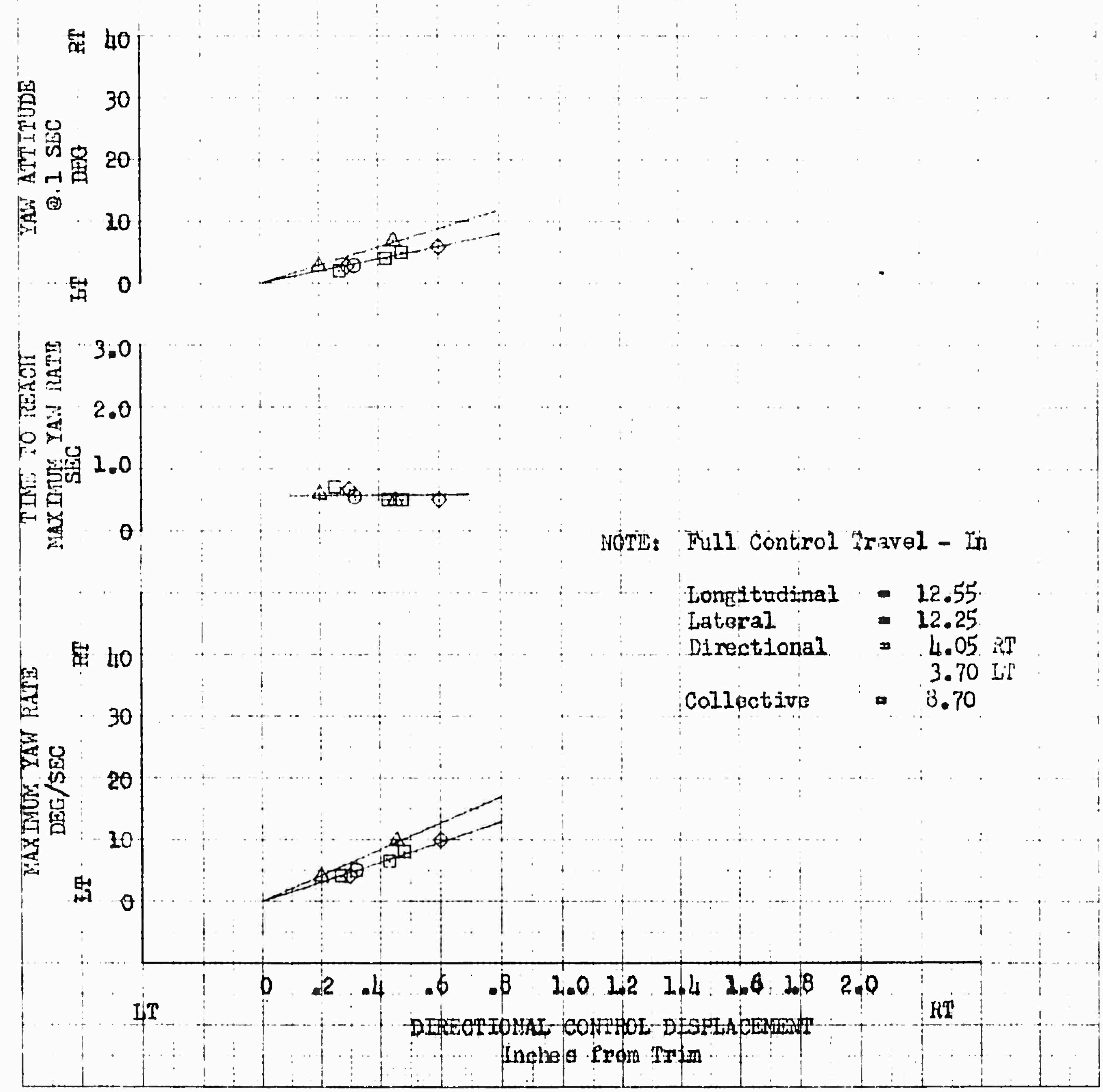


NOTE: Full Control Travel - In

Longitudinal = 12.55
Lateral = 12.25
Directional = 4.05 RT
3.70 LT
Collective = 8.70

FIGURE NO. 46
DIRECTIONAL CONTROL RESPONSE
OH-6A S/N 65-12927

SYM	CALIB. AIRSPEED KTS	DENSITY ALTITUDE FT	GROSS WEIGHT LB	ROTOR SPEED RPM	C.G LOCATION IN LONG	C.G LOCATION IN LAT	CONF.	FLIGHT CONDITION
△	81	6000	2300	483	97.2(fwd)	0(mid)	Clean	Level Flight
◇	102	6000	2290	483	97.2(fwd)	0(mid)	Clean	Level Flight
○	74	6000	2510	483	101.3(mid)	1.5(lt)	Armed	Level Flight
□	92	6000	2480	483	101.3(mid)	1.5(lt)	Armed	Level Flight



APPENDIX VI

SYMBOLS AND ABBREVIATIONS

SYMBOL	DEFINITIONS	UNIT
KTAS (V_T)	Knots True Airspeed	knots
KCAS (V_C)	Knots Calibrated Airspeed	knots
V_{NE} (V_{max})	Never Exceed Airspeed	knots
IGE	In Ground Effect	
OGE	Out of Ground Effect	
C.G.	Center of Gravity	inches
G.W.	Gross Weight	pounds
RPM	Revolutions Per Minute	
°C	Degrees Centigrade	
°F	Degrees Fahrenheit	
S.L.	Sea Level	
SHP	Shaft Horsepower	
R/C	Rate of Climb	feet per minute
C_P	Power Coefficient	
NAMT	Nautical Air Miles Travelled	
NAMPP	Nautical Air Miles Per Pound of Fuel	
N_1	Gas Producer Speed	percent RPM
N_2	Power Turbine Speed	percent RPM
H_D	Density Altitude	feet
H_P	Pressure Altitude	[illegible]
T_{t_5} (TOT)	Turbine Outlet Temperature	degrees

SYMBOL	DEFINITION	UNIT
T_{t_2}	Inlet Temperature	degrees
T_a	Ambient Temperature	degrees
P_{t_2}	Inlet Pressure	in. H_g
P_a	Ambient Pressure	in. H_g
δ	Pressure Ratio (P/P_a)	
θ	Temperature Ratio (T/T_a)	
K_W	Rate of Climb Weight Correction Factor	
K_P	Rate of Climb Power Correction Factor	

APPENDIX VII

REFERENCES

1. US Army Aviation Materiel Command (USAAVCOM), IFB No. DAAJ01-68-B-0049, General Specification for the Light Observation Helicopter (LOH), 25 July 1967.

2. Hughes Tool Company, Aircraft Division HTC-AD-369-Y-8011, Detail Specification for a Single-Engine, Single-Main-Rotor, Light Observation Helicopter Army Model Designation OH-6A, 25 September 1967.

3. Allison Division of General Motors, Model Specification No. 580-F, Amendment No. 1 Model T63-A-5A, 18 August 1965.

4. Allison Division of General Motors, Model 250 Test Data Engine, Serial Number 400105, B.V. 1, Test date, 20 September 1967.

5. US Army Aviation Test Activity, Project No. 67-13, Engineering Flight Test of the Light Observation Helicopter (LOH), October 1967.

6. US Army Test and Evaluation Command Project No. 4-3-0250/51/52/53, Engineering Flight Test of the OH-6A Helicopter, Part II, August 1964.

7. US Army Test and Evaluation Command Project No. 4-6-0250-01, Engineering Flight Test of the Production OH-6A Helicopter in the Armed and Unarmed Configuration, unpublished (data at US Army Aviation Test Activity).

8. Hughes Tool Company, Aircraft Division Project No. 4266WE-D Type Inspection Authorization, 28 September 1967.

9. Hughes Tool Company, Aircraft Division, Proposed Instrument Panel Arrangement Drawing No. 369ASK427, 20 October 1967.

10. Military Specification, MIL-C-5011A.

11. Military Specification, MIL-I-6115A.

12. Military Specification, MIL-H-8501A.

13. Weight and Balance Handbook, OH-6A, Serial Number 65-12927

APPENDIX VIII

PILOT RATING SCALE

CONTROLLABLE CAPABLE OF BEING CONTROLLED OR MANAGED IN CONTEXT OF MISSION, WITH AVAILABLE PILOT ATTENTION	ACCEPTABLE MAY HAVE DEFICIENCIES WHICH WARRANT IMPROVEMENT, BUT ADEQUATE FOR MISSION. PILOT COMPENSATION, IF REQUIRED TO ACHIEVE ACCEPTABLE PERFORMANCE, IS FEASIBLE	SATISFACTORY MEETS ALL REQUIREMENTS AND EXPECTATIONS, GOOD ENOUGH WITHOUT IMPROVEMENT CLEARLY ADEQUATE FOR MISSION.	EXCELLENT, HIGHLY DESIRABLE	A1
			GOOD, PLEASANT, WELL BEHAVED	A2
			FAIR. SOME MILDLY UNPLEASANT CHARACTERISTICS. GOOD ENOUGH FOR MISSION WITHOUT IMPROVEMENT.	A3
		UNSATISFACTORY RELUCTANTLY ACCEPTABLE. DEFICIENCIES WHICH WARRANT IMPROVEMENT. PERFORMANCE ADEQUATE FOR MISSION WITH FEASIBLE PILOT COMPENSATION.	SOME MINOR BUT ANNOYING DEFICIENCIES. IMPROVEMENT IS REQUESTED. EFFECT ON PERFORMANCE IS EASILY COMPENSATED FOR BY PILOT.	A4
			MODERATELY OBJECTIONABLE DEFICIENCIES. IMPROVEMENT IS NEEDED. REASONABLE PERFORMANCE REQUIRES CONSIDERABLE PILOT COMPENSATION.	A5
			VERY OBJECTIONABLE DEFICIENCIES. MAJOR IMPROVEMENTS ARE NEEDED. REQUIRES BEST AVAILABLE PILOT COMPENSATION TO ACHIEVE ACCEPTABLE PERFORMANCE.	A6
	UNACCEPTABLE DEFICIENCIES WHICH REQUIRE MANDATORY IMPROVEMENT. INADEQUATE PERFORMANCE FOR MISSION EVEN WITH MAXIMUM FEASIBLE PILOT COMPENSATION.		MAJOR DEFICIENCIES WHICH REQUIRE MANDATORY IMPROVEMENT FOR ACCEPTANCE. CONTROLLABLE. PERFORMANCE INADEQUATE FOR MISSION, OR PILOT COMPENSATION REQUIRED FOR MINIMUM ACCEPTABLE PERFORMANCE IN MISSION IS TOO HIGH.	U7
			CONTROLLABLE WITH DIFFICULTY. REQUIRES SUBSTANTIAL PILOT SKILL AND ATTENTION TO RETAIN CONTROL AND CONTINUE MISSION.	U8
			MARGINALLY CONTROLLABLE IN MISSION. REQUIRES MAXIMUM AVAILABLE PILOT SKILL AND ATTENTION TO RETAIN CONTROL.	U9
UNCONTROLLABLE CONTROL WILL BE LOST DURING SOME PORTION OF MISSION.			UNCONTROLLABLE IN MISSION.	10

PILOT RATING SCALE