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RDTE PROJECT NO. 1X141807D174
USATECOM PROJECT NO. 4-6-0500-01
USAASTA PROJECT NO. 66-06

ENGINEERING FLIGHT TEST

AH-1G HELICOPTER (HUEYCOBRA)

PHASE D

PART 2
PERFORMANCE

FINAL REPORT

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APRIL 1970

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

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(AIRWORTHINESS AND FLIGHT CHARACTERISTICS)

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ABSTRACT

The Phase D, Part 2 airworthiness and qualification performance tests of the AH-1G helicopter were conducted in California at Edwards Air Force Base and auxiliary test sites during the period 13 June 1968 through 29 July 1969. Specific performance parameters were evaluated to determine model specification compliance and to obtain detailed performance and mission suitability information for inclusion in technical manuals and other publications. The AH-1G exceeded all contractor performance guarantees. There were two deficiencies which affect the mission accomplishment of the helicopter: insufficient directional control which limits hovering, takeoff and landing performance; and excessive tail rotor horsepower required for hovering flight. There were three shortcomings for which corrective action is desirable: the inability to achieve maximum tail rotor blade angle (19 degrees) when full left directional control is applied for all conditions with the present directional control/yaw SCAS geometry; excessive pilot effort required to maintain optimum climb and maximum endurance airspeeds; and the possibility of inadvertently exceeding the main transmission torque limit following a left-lateral control input when below the engine critical altitude.

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INTRODUCTION

BACKGROUND

1. In October 1965, the Department of the Army directed the US Army Materiel Command (USAMC) to conduct an expedited comparative evaluation of a selected group of three helicopters to fulfill the immediate requirement for an armed helicopter. A flight test program was conducted on the three aircraft by the US Army Aviation Systems Test Activity (USAASTA) at Edwards Air Force Base (AFB), California, from 13 November to 1 December 1965. The AH-1G Hueycobra was the aircraft selected from the evaluation to meet this requirement.

2. On 17 August 1966, USAASTA was directed by the US Army Test and Evaluation Command (USATECOM) to perform Phase B and Phase D testing of the AH-1G helicopter (ref 1, app I). A plan of test for the Phase B engineering test was submitted by USAASTA in April 1967 and approved by the US Army Aviation Systems Command (USAAVSCOM). Phase B tests were conducted at different test sites and geographical locations from 3 April 1967 to 3 May 1968 on several test aircraft. The results of these tests are contained in references 2 through 8. The plan of test for the Phase D program (ref 9) was initially submitted in August 1967 and was approved by USAAVSCOM on 24 October 1968. The Phase D plan of test was amended on 5 November 1968 to include an additional test requested by USAAVSCOM (ref 10). Two aircraft were used for the Phase D test program to reduce the calendar testing time. One of the test aircraft was a prototype (aircraft serial number 6615247) and the other was a production model (aircraft serial number 6715695). The results of the Phase D performance tests are presented in this report (Part 2). The Phase D handling qualities and vibration characteristics are presented in other reports (Part 1 and Part 3). No wing store jettison or armament subsystem firing tests were conducted during the Phase D program since adequate testing had been accomplished in these areas during the AH-1G Phase B program.

TEST OBJECTIVES

3. The objectives of the AH-1G Phase D test program were:

a. To provide information for technical manuals and other service publications.

b. To determine compliance with applicable military specifications.

c. To determine compliance with contract performance guarantees.

d. To evaluate operational suitability for the armed helicopter mission.

DESCRIPTION

4. The AH-1G helicopter manufactured by Bell Helicopter Company (BHC) was designed specifically to meet the US Army requirements for an armed helicopter. Tandem seating is provided for a two-man crew. The main rotor system is a two-bladed, semirigid, "door hinge" type with the stabilizer bar removed. A conventional antitorque rotor is located near the top of the vertical stabilizer. The AH-1G is equipped with a three-axis stability and control augmentation system (SCAS) to improve the aircraft's handling qualities. The helicopter is powered by a Lycoming T53-L-13 turboshaft engine rated at 1400 shaft horsepower (shp) at sea level (SL) under standard day, uninstalled conditions. The engine is derated to 1100 shp due to the maximum torque limit of the helicopter's main transmission. Distinctive features of the AH-1G are: the narrow fuselage (36 inches), the stub midwing with four external store stations and the integral chin turret. The flight control system is of the mechanical, hydraulically boosted, irreversible type with conventional helicopter controls in the aft cockpit (pilot's station). The controls in the forward cockpit (copilot/gunner's station) consist of conventional antitorque pedals and sidearm collective and cyclic controls. An electrically operated force trim system is connected to the cyclic and directional controls to induce artificial feel and to provide positive control centering. The elevator is synchronized with the cyclic stick. The armament configurations are changed by varying the wing stores and chin turret configurations. The pilot can fire the wing stores and the chin turret only in the stowed position. The copilot/gunner operates the flexible turret and can also fire the wing stores in an emergency. The wing stores can be jettisoned by either the pilot or gunner in case of emergency. The design gross weight (grwt) for the AH-1G is 6600 pounds, and the maximum grwt is 9500 pounds. More detailed aircraft information and operating limits of the AH-1G are presented in appendix II.

SCOPE OF TEST

5. During the AH-1G Phase D test program, 256 flights were conducted for a total of 368.8 flight hours of which 227.9 hours were productive test hours. Testing was conducted in California from 12 June 1968

to 29 July 1969 at Shafter Airport (420-foot elevation), Edwards AFB (2300-foot elevation) and at high-altitude test sites near Bishop (4120-, 7010- and 9500-foot elevations). Testing was conducted to determine the aircraft performance, handling qualities and vibration characteristics. This report contains the results of the performance testing, and Part 1 and Part 3 contain handling qualities and vibration test results. Performance testing required 143.4 hours and 173 flights. All performance testing was conducted on aircraft S/N 6615247. The configurations tested during the performance portion of the program are listed in table 1.

Table 1. Aircraft Armament Configurations.

Configuration	Armament Subsystems
Clean	TAT-102A or XM28 turret, no external wing store
Basic	TAT-102A or XM28 turret, one XM157 outboard each wing
Inboard alternate	TAT-102A or XM28 turret, one XM159 inboard each wing
Outboard alternate	TAT-102A or XM28 turret, one XM159 outboard each wing
Light scout	TAT-102A or XM28 turret, one XM18 inboard each wing, one XM157 outboard each wing
Heavy scout	TAT-102A or XM28 turret, one XM18 inboard each wing, one XM159 outboard each wing
Light hog	TAT-102A or XM28 turret, two XM157 each wing
Heavy hog	TAT-102A or XM28 turret, two XM159 each wing

Note: The test aircraft was equipped with the TAT-102A chin turret: one 7.62 minigun (XM-134).

6. The test program was conducted within the limitations established by the AH-1G Safety-of-Flight Releases issued by USAAVSCOM, (refs 11 and 12, app I).

7. The empty grwt of the test aircraft in a clean configuration with test instrumentation installed was 5790 pounds with a center of gravity (cg) at fuselage station (FS) 205.97 for aircraft S/N 6615247.

8. The AH-1G was evaluated as an armed tactical helicopter, capable of day or night operation from prepared or unprepared areas. The performance of the AH-1G helicopter was evaluated to determine compliance with the requirements of paragraph 3.1.2 of the detail specification (ref 13, app I). Handling qualities ratings were assigned in accordance with the Handling Qualities Rating Scale (HQRS) presented as appendix III. Specific test conditions for each test are presented in the Results and Discussion section of this report.

METHODS OF TEST

9. Test methods and data reduction procedures used in these tests are proven engineering flight test techniques and are described briefly in appendix IV. All flights were conducted and supported by USAASTA personnel. Tests were conducted in nonturbulent atmospheric conditions so the data would not be influenced by uncontrolled disturbances.

10. The flight test data were recorded from test instrumentation in the pilot's panel, copilot/gunner's panel, photopanel and 24-channel oscillograph. A detailed listing of the test instrumentation is included in appendix V.

CHRONOLOGY

11. The chronology of the AH-1G Phase D, Part 2 test program is as follows:

Phase B flight test completed on aircraft S/N 6615247	3 May 1968
Phase D flight test commenced on aircraft S/N 6615247	13 June 1968
Flight test completed on aircraft S/N 6615247	29 July 1969
Draft report submitted	January 1970

RESULTS AND DISCUSSION

GENERAL

12. This report presents the results of the engineering Phase D performance flight tests of the AH-1G helicopter. The test data obtained during these tests were used for determining compliance with the detail specification (ref 13, app I) and to provide information for use in technical manuals and other publications. The AH-1G met or exceeded all contractor performance guarantees (see summary in app VI). There were two deficiencies which affect the mission accomplishment of the helicopter: insufficient directional control which limits hovering, takeoff and landing performance and excessive tail rotor horsepower required for hovering flight. There were three shortcomings for which corrective action is desirable: inability to achieve maximum tail rotor blade angle (19 degrees) when full left directional control is applied for all conditions with the present directional control/yaw SCAS geometry; excessive pilot effort is required to maintain optimum climb and maximum endurance airspeeds; and the possibility of inadvertently exceeding the main transmission torque limit following a left-lateral control input when below the engine critical altitude.

13. An addendum to Part 2 of the Phase D report will be published to present the test results of the turning performance, level-flight acceleration and deceleration performance and altitude loss during recovery from a dive.

AIRCRAFT CONTROL SYSTEM RIGGING

14. Prior to testing, the aircraft flight and engine controls were rigged in compliance with appropriate US Army publications. Subsequent aircraft flight and engine control rigging changes were coordinated with contractor technical representatives.

ANTITORQUE SYSTEM PERFORMANCE

15. Tests were conducted to determine the limitations of aircraft performance resulting from the antitorque system. An instrumented 90-degree tail rotor gear box was installed to measure tail rotor torque. Test data were acquired in conjunction with other tests.

16. Results of the tail rotor performance for various hovering skid heights is presented in figures 1 through 12, appendix VII. Hovering, takeoff and landing performance of the AH-1G were found to be

limited by the directional control system. Specific limitations are as follows:

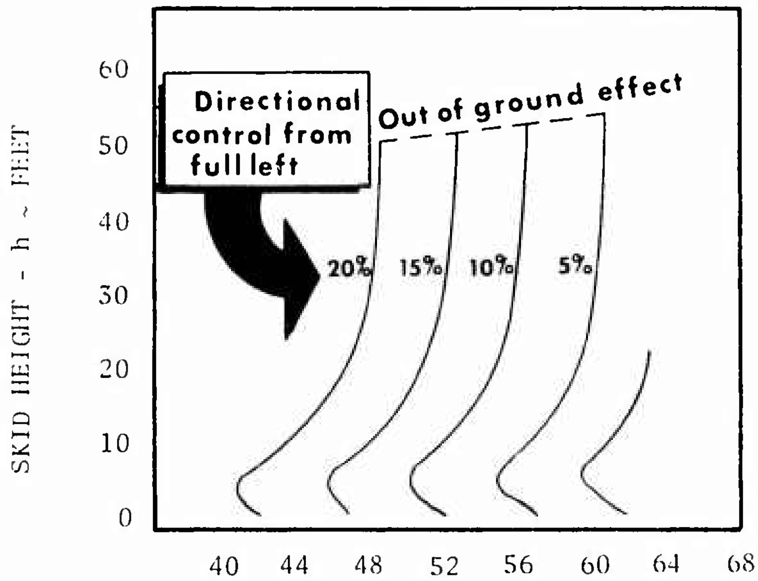
a. Insufficient directional control to maintain a desired heading.

b. Overtorquing of the tail rotor drive system.

c. Inability to attain maximum tail rotor blade angle (19 degrees) with full left directional control when the SCAS actuator is extended to the right of the null position.

17. A directional control margin of 10 percent of the full displacement while hovering was determined to be the minimum acceptable to adequately correct heading deviations caused by small wind gusts (2 to 7 knots) and small transient torque variations due to main rotor speed changes. This margin allows an 18-degree-per-second (deg/sec) left yaw rate to be generated (with the directional SCAS in the null position) when the remaining 10 percent of left directional control is applied. However, the directional control displacement limits vary as a function of directional SCAS position discussed in paragraph 20. Figure A presents the variation in directional control margin as a function of skid height. The maximum main rotor thrust coefficient (C_T) allowable for each directional control margin varied significantly with skid height between 3 and 15 feet. The skid height at which minimum main rotor C_T is obtained varied from 5 to 8 feet depending on the magnitude of the directional control margin. Above a skid height of 15 feet, the maximum C_T for a given directional control margin increased until OGE hover was attained. The influence of the lower maximum C_T attainable at specific control margins when hovering between 5 and 8 feet is discussed in Hovering Performance (paras 22 through 26), Takeoff Performance (paras 27 through 30) and Landing Performance (paras 54 and 55).

FIGURE A
DIRECTIONAL CONTROL MARGINS AS A FUNCTION OF AIRCRAFT
SKID HEIGHT IN A HOVER
AH-1G



$$\text{MAIN ROTOR THRUST COEFFICIENT-}C_T = \frac{GRWT}{\rho A(\Omega R)^2} \times 10^4$$

Notes: 1. Total Directional Control Displacement = 7.07 Inches
2. Full Left Directional Control = 19° Tail Rotor Pitch
3. Wind Less Than 2 Knots

18. The contractor's efforts to increase the directional control of the AH-1G by increasing the tail rotor blade pitch angle above 19 degrees proved to be unsatisfactory. Although the tail rotor thrust was increased, the increased torque required caused overtorquing of the tail rotor drive system components (ref 3, app I).

19. The tail rotor blades were rigged at a 19-degree ($\pm\frac{1}{4}$) maximum pitch angle with full left directional control. The horsepower

required at the output shaft of the 90-degree tail rotor gear box for various directional control margins at different hovering skid heights is presented in figures 8 through 10, appendix VII. It was found that for any given tail rotor blade angle during stabilized conditions, the tail rotor horsepower required is most critical during hovering flight. The tail rotor horsepower for a given blade angle varies as a function of density altitude (Hp), decreasing as density altitude increased. When hovering out of ground effect (OGE) with a 10-percent directional control margin, 145 shp was required at SL and 106 shp at a 10,000-foot Hp. It was also determined that when hovering at 3- to 15-foot skid heights with less than the 10-percent control margin, the tail rotor horsepower required increases nonlinearly as the directional control approaches the left limit. Although a current tail rotor drive system torque limit could not be determined, the Structural Design Criteria Report (ref 14, app I) for the AH-1G stated that the anti-torque drive system design limit was 386 foot-pounds of torque (122 shp at 1654 rpm). Analysis of the data reveals that numerous stabilized hovering flight conditions require higher tail rotor horsepower than this design point. The tail rotor horsepower encountered during translational flight or unstabilized hovering conditions are greater for many conditions than those for stabilized hover. The magnitude of tail rotor horsepower resulting from these transient maneuvers is discussed in reference 15, appendix I. During the conduct of this test program, *eight* 42-degree gear boxes and *four* 90-degree gear boxes were replaced. Any operation above 180 tail rotor shp required immediate replacement of the 42-degree gear box due to unacceptable gear wear patterns. The 90-degree gear box required replacement when operated above 180 shp for limited periods. The excessive tail rotor horsepower required and resultant drive system damage were unsatisfactory, and correction is mandatory.

20. The limits of the directional control displacement vary as a function of directional SCAS position. When the directional SCAS is nulled, full left directional control results in a 19-degree tail rotor blade pitch angle. With the directional SCAS 12.5 percent to the right of the nulled position, only a 16-degree blade angle can be attained. Thus, when operating under conditions where directional control is critical, the yaw SCAS operation can further deteriorate the maximum directional control power.

21. The following recommendations resulted from an analysis of the antitorque system performance:

a. To provide adequate directional control power and to preclude excessive overtorquing of the tail rotor drive system components, the operational flight envelope should be restricted to conditions which provide a 10-percent directional control margin. Also, hovering at skid heights of 3 to 15 feet should be avoided.

b. Action should be initiated to increase the directional control margin and improve the torque transferring capability of the tail rotor drive system so the full potential of the AH-1G can be realized.

HOVERING PERFORMANCE

22. The objective of the hovering performance tests was to determine hovering performance as a function of skid height above the ground. The tests were conducted in the clean configuration and spot-checked in the heavy hog configuration to determine the effects of wing stores. The test results are presented in figures 13 through 19, appendix VII. The test conditions are presented in table 2. Tethered hover was used as a primary test method, and the OGE data were spot-checked during free-flight hover.

Table 2. Hovering Performance Test Conditions.

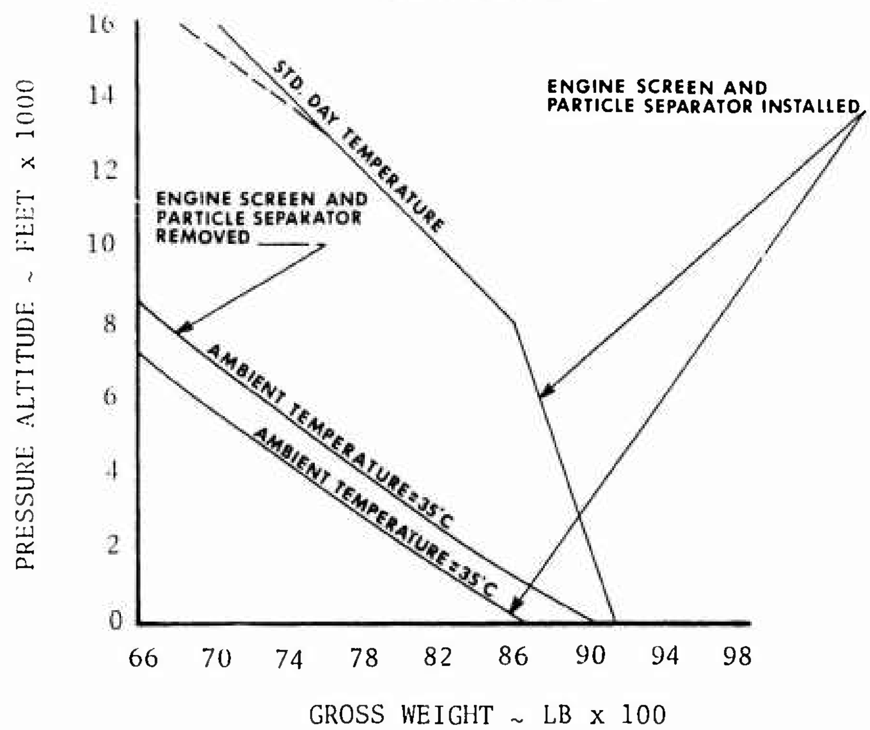
Configuration	Skid Height (ft)	Altitude Above Mean Sea Level (ft)	Rotor Speed (rpm)
Clean	IGE: 2,5,10,15,30 OGE: 100	520	324, 314
Clean and heavy hog	IGE: 2,5,10,15,30 OGE: 100	4120	324, 314
Clean and heavy hog	IGE: 2,5,10,15,30 OGE: 100	9500	324, 314

23. The AH-1G hovering performance contract guarantee states that the aircraft at an 8000-pound grwt will hover at 2000 feet OGE at an outside air temperature of 95°F. The hovering guarantee further states that the engine power available will be determined with the particle separator and engine inlet screens removed and zero bleed air extracted from the engine compressor section. Under these conditions, the aircraft exceeded the contract guarantee by 1400 feet in altitude or 430 pounds in gross weight (fig. 13, app VII). This guarantee was met without encountering the recommended 10-percent directional control margin.

24. The production aircraft has the engine particle separator and engine inlet screens installed plus an 0.6-percent compressor bleed

air extraction to drive the engine oil cooling fan. These modifications decreased the engine power available and, consequently, decreased the OGE hovering capability for an ambient temperature of 35°C. This decrease in performance is illustrated in figure B. Figure B also presents standard day, OGE hovering performance. The standard day, OGE hovering ceiling was limited by the recommended 10-percent directional control margin above 13,200 feet as indicated by the dashed line in figure B.

FIGURE B
OGE HOVERING PERFORMANCE
AH-1G
T53-L-13 ENGINE

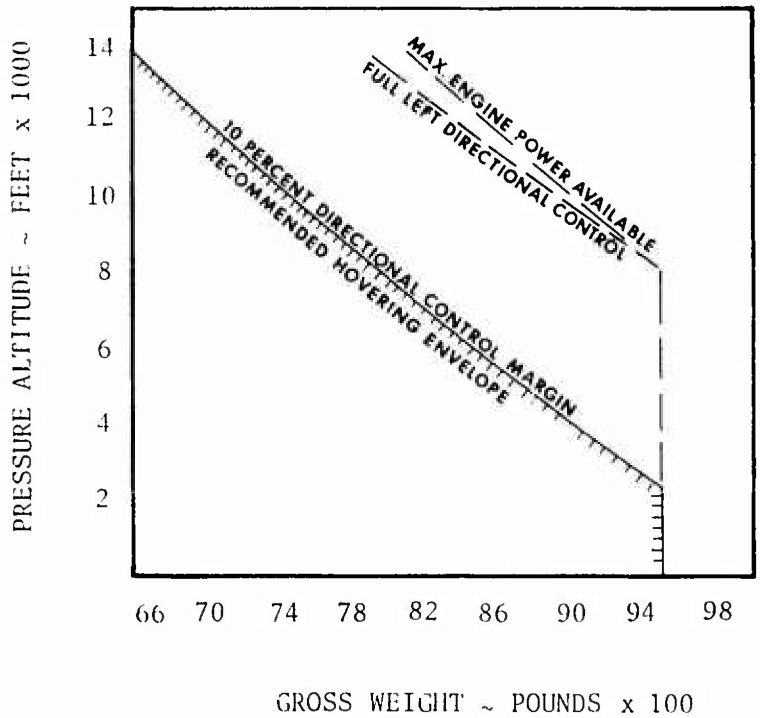


Notes: 1. Wind Less Than 2 Knots
2. Rotor Speed = 324 RPM

25. The in-ground-effect (IGE) hovering performance is limited by directional control in many areas depending on skid height, density altitude and rotor speed. The most critical IGE skid height occurs at 7 feet with a directional control margin of 10 percent. Figure C presents the IGE hovering capability of the AH-1G at a skid height of 5 feet for standard day conditions at a rotor speed of 324 rpm. It can be seen that the AH-1G hovering capability is greatly reduced when observing the recommended 10-percent directional control margin.

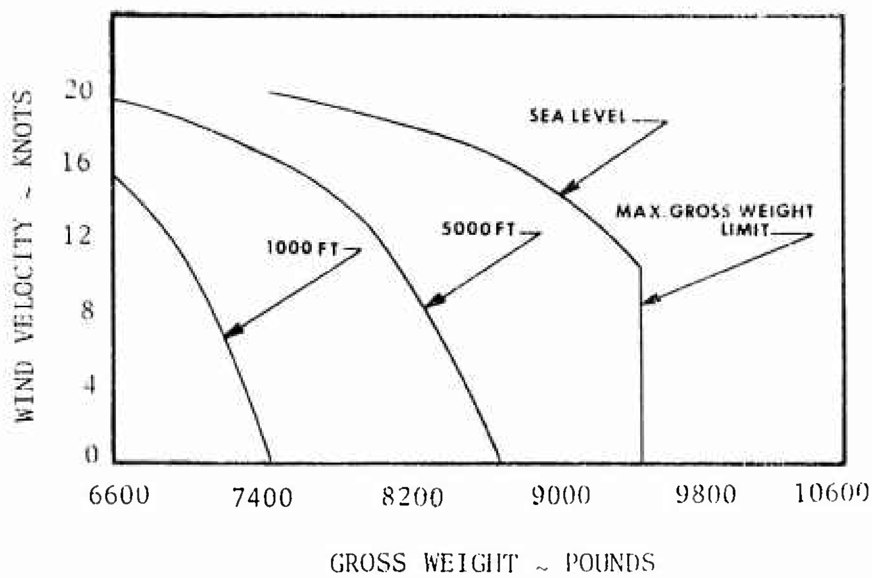
FIGURE C
IGE HOVERING PERFORMANCE
AH-1G T53-L-13
ENGINE PARTICLE SEPARATOR INSTALLED

Notes: 1. Standard Day
2. Wind Less Than 2 Knots
3. Skid Height = 5 Feet
4. Rotor Speed = 324 RPM



26. The hovering performance capability was further degraded when hovering in adverse crosswind conditions as shown in figures 20 through 25, appendix VII. Figure D is presented to illustrate this degradation. For IGE hover ceilings at 5000 and 10,000 feet, the data show that: up to and including wind velocities of 10 knots, the maximum hovering grwt is reduced approximately 55 pounds per knot; for wind velocities above 10 knots, the reduction in gross weight increases nonlinearly with increasing wind velocity at all three altitudes.

FIGURE D
HOVERING IN WIND ENVELOPE FOR A
10-PERCENT DIRECTIONAL CONTROL MARGIN
AH-1G T53-L-13
SKID HEIGHT = 7 FEET



Notes: 1. Wind Velocity Presented for Critical Wind Azimuth
2. Seven-Foot Skid Height Represents Most Critical Condition
3. Full Left Directional Control = 19° Tail Rotor Blade Angle
4. Ten-Percent Directional Control Remaining From Mean Control Position Required During Stabilized Hover
5. Yaw SCAS OFF
6. Standard Day

TAKEOFF PERFORMANCE

27. Takeoff tests were conducted to determine the takeoff distance required to clear a 50-foot obstacle. The test conditions were: heavy hog configuration, a 7420- to 9270-pound grwt, a 140- to 11,320-foot H_D and a cg of 195 inches. The test results are presented in figures 26 to 29, appendix VII. The takeoff performance summary (fig. 26) shows that takeoff performance at altitude is limited by directional control. For a 10-percent directional control margin, the maximum takeoff density altitudes attained were: 10,600 feet for a gross weight of 8500 pounds and 6600 feet for a gross weight of 9500 pounds. The performance summary does not include data below 30 knots indicated airspeed (KIAS) because of large errors in the standard airspeed system between zero and 30 KIAS. Data were obtained at airspeeds below 30 KIAS with the boom airspeed system. Although these data are included in the test results, they are not

recommended for handbook inclusion since they are neither accurate nor repeatable with the production airspeed system.

28. The test technique used during this test was as follows:

a. Hover IGE at a 3-foot skid height at 324 rotor rpm.

b. Slowly accelerate to 15 knots (translational lift) with the minimum increase in collective pitch required to maintain a 3-foot skid height.

c. Smoothly increase collective pitch to limit torque (or engine exhaust gas temperature (EGT) limit, if applicable) and continue the acceleration at the same skid height.

d. Rotate the helicopter to a climb attitude. Rotation was initiated at the climbout airspeed minus 10 percent to prevent overshooting.

e. Climbout at the recommended airspeed until clear of the obstacle.

29. The above technique differs from the normal level-flight acceleration test method where maximum power is applied at a hover. The change in technique was necessary to avoid excessive horsepower requirements in the tail rotor drive train and encountering insufficient directional control. The test takeoff technique allowed the maneuver to be performed with little or no increase in the left pedal required for a stabilized 3-foot hover. Also, this test technique, unlike the normal method, does not demand uncomfortably large, nose-down pitch attitudes to maintain a constant skid height during acceleration when performing a takeoff with high excess power available.

30. The significance of the modified technique can be seen in the time histories of this maneuver (figs. 30 and 31, app VII). These data show that the design tail rotor horsepower limit and the 10-percent directional control margin were closely approached during the initial phase of the maneuver. The data also show significant decreases in tail rotor horsepower and left pedal required with increased airspeed. For the same test conditions, the earlier power application of the normal takeoff technique would result in exceeding the tail rotor horsepower limit and reducing the directional control margin significantly. To preclude further limitations to the takeoff envelope due to either excessive tail rotor horsepower requirement or loss of adequate directional control margin, the level-flight takeoff technique used during this test is recommended for operational use.

CLIMB PERFORMANCE

31. Continuous climbs to service ceilings were conducted to determine the climb performance of the AH-1G. All climbs were performed with the engine developing 1100 shp until the critical altitude of the installed engine was reached. Above the critical altitude, military rated power (MRP) was used until service ceiling was obtained. The optimum airspeed climb schedule was used for all climbs. The test conditions and significant results for each climb are presented in table 3. It is estimated that the rates of climb presented in table 3 could be improved upon by flying at the aft cg limit. The complete test results of the continuous climbs are presented in figures 32 through 36, appendix VII. The rates of climb, particularly from SL to 10,000 feet, were excellent and enhance the capability of the AH-1G for the attack helicopter mission.

Table 3. Climb Performance Test Results.

Center of gravity: forward   Standard day
Rotor speed: 324 rpm   Rocket pod fairings not installed

Configuration	Climb Start GRWT (lb)	SL Rate of Climb (fpm)	10,000-foot H_D Rate of Climb (fpm)	Combat Ceiling[1] (ft)	Service Ceiling[2] (ft)
Clean	7500	2200	2150	19,500	20,900
Clean	8500	1725	1625	16,600	18,100
Heavy hog	8500	1675	1550	16,500	17,900
Heavy hog	9500	1250	1050	12,600	14,200

[1]Altitude for maximum rate of climb of 500 feet per minute (fpm).
[2]Altitude for maximum rate of climb of 100 fpm.

32. The climb performance contract guarantee states that the aircraft will climb at 1800 fpm on a standard day at SL in the outboard alternate configuration with a climb start grwt of 8000 pounds. Due to atmospheric conditions and the altitude specified by the contract guarantee, it was necessary to extrapolate the test data from 2400 feet to SL. The extrapolation indicates a SL rate of climb (R/C) of 1835 fpm (35 fpm more than required by the guarantee). It is estimated that an additional 65-fpm R/C could be realized by flying at an aft cg loading.

33. Additional continuous climbs were flown from 2000 to 10,000 feet to determine the correction factors for both gross weight (K_W) and engine power (K_p) changes. These climbs were conducted in both the clean and heavy hog configurations. A value of 0.873 was determined for K_p in both configurations. K_W varies nonlinearly from 0.560 for a gross weight of 7000 pounds to 1.026 for a gross weight of 9500 pounds. Altitude had no effect on the values of either K_p or K_W. The results of these tests are presented in figure 35, appendix VII.

34. The maximum R/C airspeed schedules were derived from the level-flight performance data and are presented in nondimensional form in figure 36, appendix VII. The pilot's effort required to fly the climb schedule was moderate in that numerous longitudinal control corrections were necessary to maintain an exact airspeed (HQRS 4). A reduction in pilot effort was realized by flying a climb airspeed approximately 15 knots faster than the optimum airspeed. Climbs performed at the higher airspeed resulted in satisfactory climb performance with minimal pilot compensation (HQRS 3). It is recommended that the optimum climb airspeed be increased 15 knots for night operations or instrument flight.

LEVEL FLIGHT PERFORMANCE

35. The objectives of these tests were to define level-flight maximum airspeeds and to determine optimum cruise airspeeds for maximum range and endurance. The conditions tested are presented in table 4.

Table 4. Level-Flight Test Conditions.

Configuration	Center of Gravity	Thrust Coefficient Range
Clean	Forward	0.003823 to 0.006664
Clean[1]	Forward	0.004650 to 0.005382
Clean	Aft	0.004298 to 0.006667
Basic	Forward	0.004613 to 0.005319
Basic[2]	Forward	0.004661 to 0.005419
Light scout	Forward	0.004562 to 0.005371
Light hog	Forward	0.004564 to 0.005548
Inboard alternate	Forward	0.004630 to 0.005351
Outboard alternate	Forward	0.003988 to 0.005346
Heavy scout	Forward	0.004576 to 0.006717
Heavy hog	Forward	0.003983 to 0.006676
Heavy hog[2]	Forward	0.004976 to 0.005735
Heavy hog	Aft	0.004624 to 0.006734

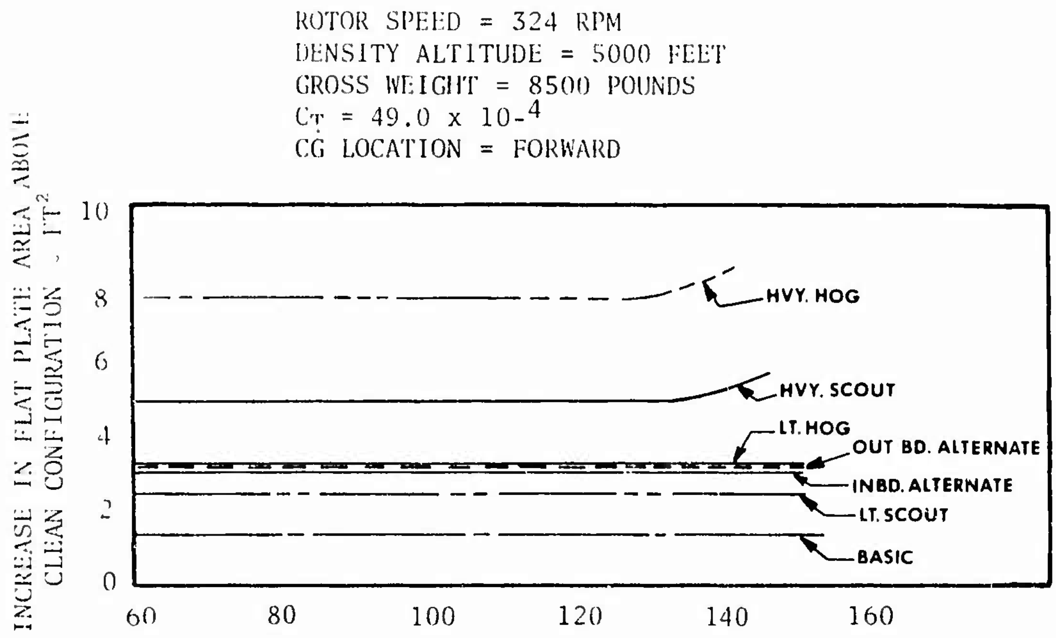
[1]Landing gear cross-tube fairings removed.
[2]Rocket pod fairings installed.

36. All tests were flown with the frangible rocket pod fairings removed unless otherwise specified. End plates were placed over the front of each rocket pod to aerodynamically simulate a loaded pod when inert rockets were not used to achieve the desired aircraft loading. The results of the level flight test are presented in figures 37 through 101, appendix VII. Aircraft endurance, specific range and maximum airspeed in level flight (V_H) for minimum and maximum aerodynamic drag are summarized in figures 106 through 109.

37. All configurations tested revealed an increase in equivalent flat plate area when compared to the clean configuration. The increase in equivalent flat plate area for different configurations is presented in figure E for a thrust coefficient of 49.0×10^{-4}.

The increase in equivalent flat plate area was greatest for the heavy scout and heavy hog configurations. The equivalent flat plate areas for these two configurations increased nonlinearly at higher airspeeds. This nonlinear increase in equivalent flat plate area was attributed to the change in aircraft attitude (nose down) as airspeed increased.

FIGURE E
CHANGE IN EQUIVALENT FLAT PLATE AREA
DUE TO WING ARMAMENT CONFIGURATION CHANGES



38. The V_H contract guarantee is 144 knots true airspeed (KTAS) in the outboard alternate configuration on a standard day at SL for a gross weight of 8000 pounds with the engine developing 1100 shp. The model specification did not specify what cg would be used to meet this guarantee or any other contract guarantee. Figure 102, appendix VII, presents the results of the V_H contract guarantee check. The aircraft did not meet this guarantee at the forward cg location since it could only achieve a velocity of 140 KTAS. However, the aircraft exceeds the contract guarantee by 9 knots when loaded at an aft cg loading.

39. The V_H was limited by the main transmission torque limit up to the critical altitude of the engine. Above the critical altitude, maximum engine power available was the limiting parameter. At 5000 feet, the V_H decreased from 154 KTAS at a 7000-pound grwt to 142 KTAS at a 9500-pound grwt in the clean configuration at a forward cg. The V_H for each individual armament configuration is presented in table 5. When comparing the clean and heavy hog configurations, the V_H decreased about 9.0 percent. The present limit airspeed (V_L) cannot be exceeded under any level flight condition. Figure F presents the maximum airspeed obtainable versus gross weight for the clean and heavy hog configurations at the forward and aft cg.

FIGURE F
MAXIMUM AIRSPEED IN LEVEL FLIGHT
AH-1G T53-L-13
ALTITUDE = 5000 FEET
ROTOR SPEED = 324 RPM
STANDARD DAY
ROCKET POD FAIRINGS REMOVED

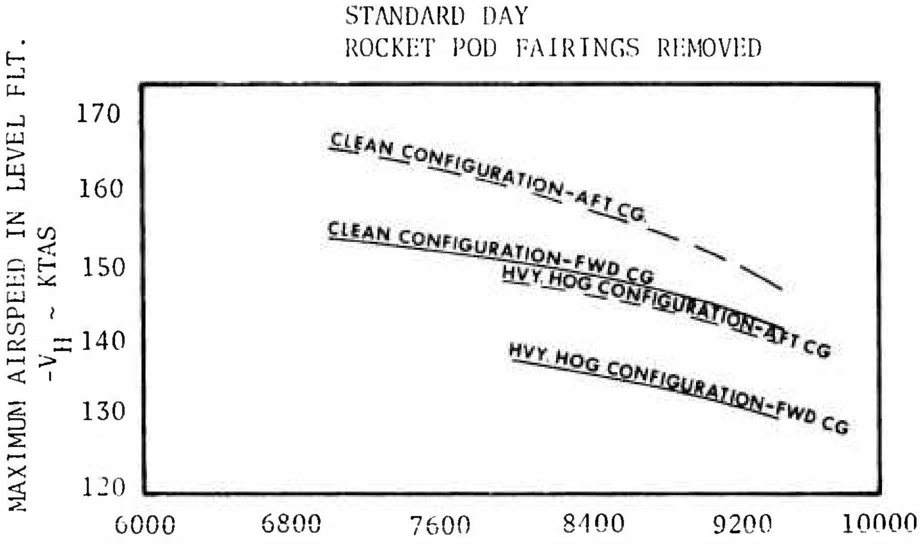


Table 5. Summary of Maximum Airspeed, Specific Range and Endurance.

Altitude: 5000 feet — Rotor speed: 324 rpm
Gross weight: 8500 pounds — Standard day
Center of gravity: forward — Rocket pod fairings not installed

Configuration	Maximum Airspeed in Level Flight (KTAS)	Recommended Cruise Airspeed for 99% Maximum NAMPP[1] (KTAS)	Specific Range for 99% Maximum NAMPP	Fuel Flow at Minimum Engine Power Required (lb/hr)	Minimum Power Required Airspeed (KTAS)	Change in Equivalent Flat Plate Area - Δf^2 (ft^2)
Clean	148.5	137.0	0.2200	458	72.0	0
Basic	146.5	135.5	0.2165	462	70.5	1.4
Light scout	144.5	134.0	0.2140	464	69.5	2.5
Inboard alternate	143.5	133.0	0.2130	464	68.0	3.1
Outboard alternate	143.0	132.0	0.2125	466	67.5	3.2
Light hog	142.5	131.0	0.2120	466	67.0	3.3
Heavy scout	140.5	129.5	0.2090	467	66.5	4.95 below 130 KTAS
Heavy hog	135.5	127.0	0.2030	471	63.5	7.7 below 125 KTAS

[1]Nautical air miles per pound of fuel.
[2]Δf equals equivalent flat plate area for configuration minus equivalent flat plate area for clean configuration.

40. The range performance contract guarantee states that the aircraft will have an operating radius of 148 miles. The AH-1G exceeds this contract guarantee by 1.6 nautical miles (NM) at a forward cg

and 8.1 NM at an aft cg. Table B, appendix VI, presents a summary of the operating radius contract guarantee analysis. This analysis was based on figures 103 and 104, appendix VII.

41. The range performance of various armament configurations are presented in table 5 for a thrust coefficient of 49.0×10^{-4}. The range of the heavy hog configured aircraft at 99 percent of maximum NAMPP is 7.7-percent less than the range for the clean configured aircraft. Minimal pilot effort was required to maintain the cruise airspeeds for all configurations (HQRS 3).

42. The endurance guarantee specified that the aircraft would be capable of loitering in level flight for a period of 3.0 hours. The aircraft exceeded this guarantee by 0.03 hours at a forward cg and 0.08 hours at an aft cg. Table C, appendix VI, presents a summary of the endurance contract guarantee analysis. This analysis is based on figures 103 and 104, appendix VII.

43. The endurance capability of the AH-1G in various armament configurations is presented in table 5 for a thrust coefficient of 49.0×10^{-4}. The aircraft's endurance in the minimum aerodynamic drag configuration is 2.8 percent more than in the maximum aerodynamic drag configuration.

44. Extensive pilot compensation was necessary to precisely maintain the airspeed for maximum endurance (HQRS 6). The compensation required was both annoying and fatiguing to the pilot, particularly for periods of time in excess of 15 minutes. If this airspeed was not precisely maintained, a rate of descent (R/D) developed which necessitated an increase in power to return to level flight. The pilot's effort decreased significantly (HQRS 3) while maintaining an airspeed approximately 15 knots higher than the maximum endurance airspeed. The increase in engine power required to maintain the higher airspeed was small and resulted in a maximum 3-percent increase in fuel flow. It is recommended that a discussion of the pilot's workload versus the aircraft's maximum endurance capabilities be included in the operator's manual.

45. The cg location had a significant effect on the power required for airspeeds above 50 KCAS. The power required to maintain level flight decreased as the cg moved aft. There was a larger reduction in power required for the heavy hog than the clean configuration. The reduction in equivalent flat plate area and the increase in maximum airspeed, endurance and range due to the change in cg are presented in table 6. This analysis indicates that a greater reduction in power required can be realized by operating at an aft cg.

Table 6. Effect of CG on Maximum Airspeed, Specific Range and Endurance.

Altitude: 5000 feet
Gross weight: 8500 pounds
Rotor speed: 324 rpm

Standard day
Rocket pod fairings not installed

Configuration	Maximum Airspeed in Level Flight (KTAS)	Recommended Cruise Airspeed for 99% Maximum NAMPP (KTAS)	Specific Range for 99% Maximum NAMPP	Fuel Flow at Minimum Engine Power Required (lb/hr)	Minimum Power Required Airspeed (KTAS)	Change in Equivalent Flat Plate Area - Δf[1] (ft^2)
Clean[2]	149.0	137.0	0.2200	458	72.0	3.9
Clean[3]	157.5	150.0	0.2325	455	69.0	
Heavy hog[2]	137.5	127.0	0.2030	471	63.5	6.4
Heavy hog[3]	146.0	132.5	0.2180	458	69.0	

[1]Δf equals equivalent flat plate area for forward cg minus equivalent flat plate area for aft cg.
[2]Forward cg.
[3]Aft cg.

46. The effects of main rotor compressibility were checked; however, the limited temperature range available during the test program was only sufficient to achieve a blade tip mach number change of 0.014 at 140 KTAS. Figure 105, appendix VII, presents a comparison of the blade tip mach number for two speed-power polars flown at an average thrust coefficient of 60.35×10^{-4}. This limited check indicated no significant degradation in the level flight performance with increasing tip mach numbers.

47. The installation of the frangible rocket pod fairings reduced the engine power required to maintain level flight. This reduction in power was greatest for the heavy hog configuration. The decrease in engine power required with the fairings installed was less significant in the basic configuration than in the heavy hog configuration. Table 7 presents the decrease in equivalent flat plate area and subsequent increases in maximum airspeed, endurance and range with the frangible rocket pod fairings installed for a thrust coefficient of 49.0×10^{-4}.

Table 7. Effect of Frangible Rocket Pod Fairings on Maximum Airspeed, Specific Range and Endurance.

Gross weight: 8500 pounds   Rotor speed: 324 rpm
Altitude: 5000 feet   Standard day
Center of gravity: forward

Configuration	Maximum Airspeed in Level Flight (KTAS)	Recommended Cruise Airspeed for 99% Maximum NAMPP (KTAS)	Specific Range for 99% Maximum NAMPP	Fuel Flow at Minimum Engine Power Required (lb/hr)	Minimum Power Required Airspeed (KTAS)	Change in Equivalent Flat Plate Area - Δf[1] (ft^2)
Basic[2]	146.5	135.5	0.2165	462	70.5	0.5
Basic[3]	147.5	136.5	0.2175	460	70.5	
Heavy hog[2]	137.5	127.0	0.2030	471	63.5	3.8
Heavy hog[3]	141.0	131.0	0.2100	457	68.0	

[1] Δf equals equivalent flat plate area for rocket pod fairings not installed minus equivalent flat plate area for rocket pod fairings installed.
[2] Rocket pod fairings not installed.
[3] Rocket pod fairings installed.

48. The removal of the landing gear cross-tube fairings increased the equivalent flat plate area by 0.5 square feet. This increase in the flat plate area caused a decrease of less than 2 percent in range performance and maximum level-flight airspeed. There was a negligible effect on endurance capability. A V_L of 160 KIAS for this configuration was established by USAAVSCOM's message (ref 16, app I) and was not exceeded during level-flight testing.

49. The tail rotor horsepower required was monitored during several level-flight performance tests. This parameter does not limit the operational, forward level-flight envelope. The tail rotor horsepower in forward flight above 40 knots calibrated airspeed (KCAS) varied from 15 to 45 horsepower. The higher values were encountered at maximum airspeed.

50. The equivalent flat plate area of both the test aircraft and the production AH-1G have been increased approximately 5 square feet above that of the Bell Helicopter Company's model 209 aircraft (ref 17, app I). It should be noted that the engine used during the Army evaluation of the Bell model 209 was not calibrated below an output torque pressure of 44.5 pounds per square inch (psi); therefore, this increase in equivalent flat plate area can only be calculated accurately at engine shp above 1020. This increase in equivalent flat plate area was probably caused by the following external changes:

a. The addition of two inboard wing stores stations.

b. The wider fuselage configuration for acceptance of the final chin turret.

c. Increased thickness of the stub wings.

d. Different configurations of the skid tubes and supporting structure.

e. The removal of flush-head rivets from the tail boom.

f. The addition of various access and vent panels.

AUTOROTATIONAL DESCENT PERFORMANCE

51. Steady state autorotational descent performance tests were conducted in both the clean and heavy hog configuration under test conditions of: a 5000-foot H_D, an 8500-pound grwt and a forward cg location. The test results are presented in figures 110 and 111, appendix VII. The minimum R/D was 1815 fpm for both configurations and occurred at 77.5 KTAS in the clean configuration and 74 KTAS in the heavy hog configuration. The data also indicate that airspeed can vary ±10 knots from the minimum R/D airspeed without significantly increasing R/D. This is a desirable characteristic since it allows the pilot to concentrate on such things as the landing site selection without incurring a large penalty should the airspeed vary as much as ±10 knots from the optimum.

52. The airspeed for maximum glide distance in the clean configuration was 112 KTAS and resulted in a 2140-fpm R/D and a glide ratio of 5.2:1. For the heavy hog configuration, the airspeed for maximum glide was 98 KTAS with a 2015-fpm R/D and a 4.9:1 glide ratio. Minimal pilot effort was necessary to maintain the airspeeds for maximum glide (HQRS 3).

53. Precise control of the rotor speed during steady state autorotation was difficult because small adjustments of the collective pitch control resulted in relatively large changes in rotor rpm. In addition, the high inertia of the rotor system caused a lag in the response of rotor speed to collective control inputs. These two characteristics resulted in the pilot's tendency to "chase the rotor speed". Although it was not difficult to maintain rotor rpm between red lines (294 and 339 rpm), attempting to maintain a precise rotor speed required extensive pilot effort and attention (HQRS 6).

LANDING PERFORMANCE

54. Landing performance tests were conducted to determine the landing distance required to clear a 50-foot obstacle. The test was performed at a 6360-foot H_D, at gross weights from 8490 to 9500 pounds and in the heavy hog configuration.

55. The test results are presented in figure 112, appendix VII. The slowest recommended approach airspeed is 15 KCAS (17 KTAS) and resulted in a landing distance of 265 feet after clearing a 50-foot obstacle. Although the data show that slower approach airspeeds were flown, airspeeds below translational lift (less than 15 KCAS) are not recommended because of the critical tail rotor horsepower requirements and directional control margins previously discussed (paras 17 and 19). The following landing technique was used for this test:

a. Establish the selected approach airspeed with an approximate 300-fpm R/D.

b. Maintain airspeed and R/D until the helicopter is 10 to 15 feet above the terrain.

c. Smoothly reduce airspeed and R/D and affect touchdown with little or no ground speed.

ENGINE INSTALLATION LOSSES

56. The objective of these analyses and tests was to determine engine installation losses and their effect on engine power and engine fuel flow. Engine power available and fuel flow were derived by the methods presented in the engine manufacturer's model specification (ref 18, app I). The engine power and fuel-flow data are presented in figures 114 through 118, appendix VII.

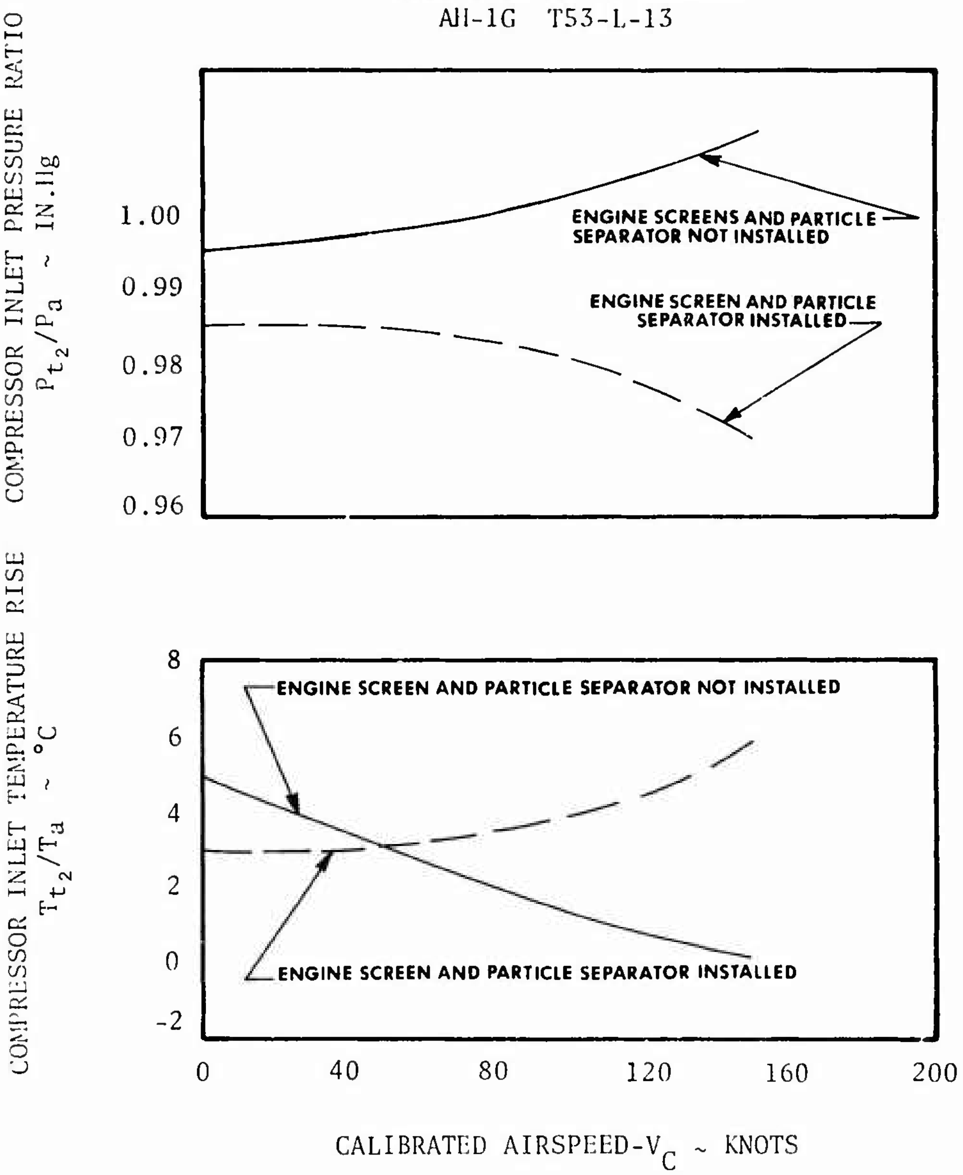
57. Only one inlet configuration was tested to determine the effect on engine performance. This inlet configuration had the engine inlet screens and engine particle separator installed and is the standard production configuration. The inlet losses attributed to this inlet configuration were used to determine engine horsepower output and engine fuel flow in all tests in this report except for contract guarantee compliance checks. The results of the production inlet evaluation are presented in figure 113, appendix VII.

58. The engine inlet temperature rise for the standard production configuration varied nonlinearly from 3°C to 6°C for airspeeds from zero to 150 KCAS. There was no apparent change in inlet temperature rise as a function of hover skid height, accumulative hover time, engine power required, altitude or rotor speed. However, a slight increase in inlet temperature can be expected when hovering down wind. The exact magnitude of this additional inlet temperature is transient in nature. No tests were conducted to determine the effects of dirt and debris accumulation in the particle separator and engine screens. The variation of inlet temperature rise had the same effect on both engine power and fuel flow which results from an increase in ambient temperature.

59. The engine inlet pressure ratio (P_{t_2}/P_a) varies nonlinearly from a maximum of 0.985 at zero airspeed (hover) to 0.97 at 150 KCAS with the standard production inlet configuration. The inlet pressure ratio did not vary with hover skid height, accumulative hover time, engine power required, altitude or rotor speed. This decreasing inlet pressure ratio with increasing airspeed caused a loss in engine power only and did not significantly affect specific fuel consumption.

60. Where applicable, all contract guarantees were based on the inlet characteristics presented in reference 17, appendix I. Confirmation of these inlet characteristics was not necessary since all contract guarantees were exceeded using these inlet losses as a data basis. This inlet configuration did not have engine inlet screens or an engine particle separator installed. A comparison of the two inlet configurations is presented in figure G.

FIGURE G
ENGINE CHARACTERISTICS
AH-1G T53-L-13



61. The magnitude of extracted compressor bleed air used was not measured during this test program. The results and analysis of the test conducted by the airframe manufacturer revealed that a constant 0.6 percent of the total air flow is used to power the aircraft's oil cooler fan. This value was used to determine engine performance when bleed air was not used for the environmental control system, engine anti-icing or rain removal. When the cooling portion of the environmental control system was operating, 3.6 percent of engine bleed air was used to determine the performance of the engine. For normal operations, bleed air extraction will probably not exceed 3.6 percent. It would be possible to use the entire 4 percent of maximum allowable bleed air if the anti-icer and cooling systems were functioning at the same time. Zero bleed air was used for checking contract guarantees.

62. Power extracted from the gas producer section varied between zero and 14.0 horsepower depending upon the electrical load. The analysis in this report including contract guarantees assumed zero horsepower extracted from the gas producer. A 14.0-horsepower extraction resulted in a maximum decrease of 1 percent in engine power available and an increase of 0.05 percent in fuel flow for a standard day.

POWER AVAILABLE

63. The objective of these analyses is to present the engine military power available as a function of airspeed, altitude and ambient temperature. The installation losses discussed previously in paragraphs 58 through 62 were used in determining engine power available. Constant values were assumed for horsepower extracted from the gas producer (zero horsepower) and power turbine output speed (6600 rpm). Power available was calculated using zero, 0.6 and 3.6 percent of engine bleed air. The power available data are presented in figures 114 through 118, appendix VII.

64. The characteristics of the production engine air inlet caused the power available to decrease with increasing airspeed. The decrease in power available was approximately 4.5 percent between zero and 160 KTAS at 10,000 feet.

65. The contract guarantees were based on the inlet characteristics presented in reference 17, appendix I. The bleed air and horsepower extracted from the compressor section were assumed to be zero since the contract guarantees did not specify any value.

66. The T53-L-13 engine is rated at 1400 shp at SL, standard day, uninstalled conditions. The maximum power output limit below the

critical altitude of the engine is defined by various contractor documents (refs 13 and 14, app I) and the US Army AH-1G operator's manual (ref 19). The maximum power output limit varies from 1100 to 1158 shp at a rotor speed of 324 rpm (6600 rpm engine power turbine) depending on which reference is used to define the limit. All performance data in this report are based on an 1100-shp limit up to the critical altitude of the engine. The variation in maximum power output limits is presented in table 8. The AH-1G operator's manual presents a "redline" engine torque limit of 50 psi in chapter seven, while 1100 shp is defined as 49.0 psi in chapter fourteen. These values disagree with the torque limits presented in table 8. It is recommended that the AH-1G operator's manual be corrected throughout to reflect a compatible engine torque limit.

Table 8. Maximum Power Output Determinations.

Engine speed: 6600 rpm
Standard day

Engine Critical Altitude (ft)	Shaft Horsepower Available (shp)	Engine Output Torque (ft-lb)	Engine Torque Pressure (psi)	Source of Information
8200	[1]1100	875	47.5	Ref 13, app I
7000	1137	[2]905	49.1	Ref 14, app I
6300	1158	921.5	[3]50.0	Ref 19, app I

[1]Engine power rating limit.
[2]Main transmission input torque limit.
[3]Engine "redline" torque pressure limit.

ENGINE CHARACTERISTICS

67. The objectives of these tests were to evaluate engine/airframe matching characteristics and to compare the contractor's engine calibration data with the engine data obtained from this test program.

68. The engine's static "droop" characteristics were good. Few adjustments were required on the power turbine speed-select "beep"

switch when reducing or increasing engine power output. The engine power turbine speed-select switch characteristics are presented in figure 119, appendix VII. The average time required for rotor speed to change after the "beep" switch was activated was 0.65 seconds. There was no noticeable variation in this delay time between a loaded or unloaded rotor system. The engine "beep" switch trim rate had a constant value of 157 rpm/sec after the delay time. The power turbine speed-select switch characteristics were satisfactory and much improved over previous UH-1 series aircraft equipped with the T53 series engine (HQRS 3).

69. The dynamic characteristics of the T53-L-13 appeared to be satisfactory throughout the flight envelope tested. When rapid power demands were required, compressor stall was not encountered during engine acceleration. Power overshoot was small and engine oscillations damped quickly.

70. A slight engine oscillation was noted when operating the engine at maximum power available above the critical altitude of the engine. This oscillation was not as serious as that reported in reference 20, appendix I. The engine oscillation was eliminated when power was reduced slightly below the maximun available.

71. Tests were performed to further define an engine-airframe-matching shortcoming previously reported in the AH-1G Phase B reports (refs 2 and 8, app I). This shortcoming was the increase in engine power output resulting from a rapid left-lateral control input while in forward flight. Conversely, a right-lateral control input resulted in a reduction in engine power output. With fixed collective and directional controls, a rapid left-lateral control input caused a decrease in rotor speed. The engine's power-turbine governor sensed the reduced rotor rpm and increased the fuel flow, thus increasing engine power output. The test data are presented in figures 120 through 124, appendix VII, and indicate that the amount of increased engine power is a function of the size of the lateral control input. Engine torque increased 5 psi as a result of left-lateral control inputs of approximately 1.5 inches at 67 and 125 KCAS. A 14-psi torque increase was recorded for a 4-inch left-lateral control input at 108 KCAS. When operating below engine critical altitude, an abrupt left-lateral cyclic input could result in exceeding the torque limit of the main transmission. This shortcoming detracts from the overall mission effectiveness of the AH-1G, and correction is desirable. Until such correction is accomplished, it is recommended that a complete discussion of this engine-airframe characteristic be included in the operator's manual.

72. Referred engine parameters were monitored throughout the test program to check for engine degradation as a function of usage.

Two engines (S/N LE 14001 and S/N LE 14008) were used during the program. The engine referred parameters calculated for these tests are presented in figures 125 through 130, appendix VII.

73. The S/N LE 14001 engine was used for all tests during the program for which engine power was required as a primary parameter. Correlation was very good between engine referred parameters obtained during this program and the engine calibration referred parameters for both the pre-program and post-program engine calibrations. The characteristics of this engine were better than the minimum acceptable standards specified in reference 18, appendix I. The only area where there was a marked difference between the engine calibration information and test program data was in the referred parameters for engine EGT. A total of 225.75 engine operating hours was accumulated during the test program. The only change that could be construed to constitute engine deterioration was a slight increase in referred EGT as a function of referred shp when comparing the pre-program and post-program calibration.

74. A total operating time of 56.2 hours was flown on engine S/N LE 14008 prior to its failure. The failure was noted during a routine preflight inspection. Visual inspection revealed that the power turbine wheel was rubbing against the casing.

ENGINE RESTART DURING FLIGHT

75. Tests were conducted to determine: the feasibility of attempting engine restarts in flight and, if practical, the best procedure to follow; altitude loss during a restart; and the engine/aircraft handling characteristics during the restart. Three engine restarts were performed during flight, two at a 5000-foot H_D and one at a 12,000-foot H_D. The first restart was made from a steady autorotation at 65 KIAS at 5000 feet using the procedure outlined in paragraph 4-26 of the AH-1G operator's manual. The second was at 110 KIAS at 5000 feet using the normal engine start procedure (governor switch in AUTO). The third was at 60 KIAS at 12,000 feet using the normal engine start procedure.

76. The results of the restart tests show that it is possible to restart the engine during flight if time and altitude are available. Following engine shutdown, the compressor speed (N_1) decayed rapidly and showed no tendency to continue rotation due to inlet airflow. The decay was not noticeably affected by different airspeeds (65 and 110 KIAS). The EGT remained high (380°C) for more than 60 seconds of flight with the engine off. Engaging the starter caused the SCAS to disengage due to low voltage. The SCAS disengagement resulted in a distracting trim change during the time when

close monitoring of N_1 and EGT was required. Due to the high residual EGT at start initiation, close monitoring and engine control were required to prevent engine overtemping. The engine acceleration capability was limited by high EGT (700°C to 760°C). The control of the engine and rotor speed (during the transition from power-off to powered flight) with the EMER governor control demanded very close attention which left little opportunity for evaluation of potential landing sites or ground proximity. The procedure using the EMER governor control is unacceptable for any situation other than a known governor malfunction. The compounding problems (engine failure, SCAS disengagement, EMER governor control) are too demanding to expect safe recovery to powered flight. Sixty seconds were required after engine starter engagement to regain sufficient power to arrest the descent. The altitude loss was 1800 feet in stablized autorotation at the minimum R/D airspeed (65 KIAS).

77. The second restart was initiated at 5000 feet in full autorotation at 110 KIAS. The governor switch was left in AUTO, and the throttle was positioned in the normal start detent. The N_1 decay following shutdown was similar to that at 65 KIAS. The EGT remained high (380°C to 390°C) at time of starter engagement. The SCAS disengaged with starter activation but was less distracting since the resultant aircraft trim change was anticipated. The EGT required close monitoring, and some throttle movements were necessary to prevent the EGT from exceeding 750°C. After self-sustaining rpm was reached (40-percent N_1), the EGT was easily controlled, and the engine accelerated smoothly to operating rpm. The time required to regain powered flight was 45 seconds. This was less time than that required when using manual throttle control of the governor (EMER). The altitude loss was about the same (1800 feet) due to the higher R/D at the higher airspeed. The pilot's attention required in the cockpit using AUTO governor control was greatly reduced from that using the EMER governor control, and the restart time was significantly reduced.

78. The third restart was made at 12,000 feet in a 60-KIAS autorotation. The AUTO governor position was used. This restart was identical to the second in all respects except for altitude loss (ie, handling qualities, pilot workload and time to restart). The altitude loss was 1350 feet.

79. The results of the test indicate that paragraph 4-27 of the AH-1G operator's manual should be revised as follows:

> 4-27. The conditions which would warrant an attempt to restart the engine would be: an engine flameout analyzed to be a malfunction of the fuel

control unit; failure of the boost pump or full closure of the throttle due to flight idle stop failure. The decision to attempt an engine restart during flight is the pilot's responsibility and is dependent upon analysis of: the cause of failure, the altitude and time available, the potential landing condition sites and the crew assistance available. Tests have shown that 45 to 60 seconds will be required to regain powered flight from the time the starter switch is depressed. Depending on the aircraft's weight, speed and flight path at the time of failure, altitude loss during restart will vary between 1500 and 2000 feet. Before making a decision, the pilot should analyze the following variables: the time and altitude required following the engine failure to regain aircraft control, the cause of failure and whether or not to set the controls and switches for restart. If an engine restart is to be attempted, proceed as follows:

> WARNING
>
> DUE TO THE INCREASED ELECTRICAL LOAD ON THE BATTERY, THE SCAS WILL DISENGAGE WHEN THE STARTER IS DEPRESSED. BE PREPARED FOR AN AIRCRAFT TRIM CHANGE.

a. Establish autorotation and select a landing area.

b. Analyze cause of failure:

(1) Mechanical: DO NOT ATTEMPT RESTART

(2) Fuel starvation: Due to throttle being closed, fuel switched OFF or boost pump failure, use abbreviated normal start procedure:

Battery....................... ON

Fuel Switch.................. ON

Boost Pump Circuit Breakers.... IN

Starter and Igniter Circuit Breakers......................	IN
Throttle	IN DETENT
Starter.......................	PULL ON, HOLD UNTIL ENGINE IS SELF SUS-TAINING AT 40-PERCENT N_1
EGT and N_1....................	MONITOR AND CONTROL WITH THROTTLE UNTIL OPERATING RPM IS RE-ESTABLISHED

(3) Fuel starvation while operating in GOV EMER:

Throttle......................	OFF
Governor Switch................	EMER
Fuel Switch...................	Check ON
Battery Switch................	Check ON
Boost Pump Circuit Breakers....	Check IN
Starter and Igniter Circuit Breakers......................	Check IN
Starter.......................	PULL ON AND HOLD
Throttle......................	OPEN SLOWLY WHEN N_1 REACHES 10 PER-CENT, CONTROL RATE OF OPENING TO KEEP EGT BELOW START LIMITS WHILE MAIN-TAINING A SMOOTH INCREASE IN N_1
Starter.......................	RELEASE WHEN EN-GINE IS SELF SUS-TAINING, 40-PERCENT N_1

Throttle......................	CONTINUE TO OPEN SLOWLY UNTIL OPERATING RPM IS REACHED. MONITOR N_2 CLOSELY AS POWERED FLIGHT IS RE-ESTABLISHED TO PREVENT ENGINE AND ROTOR OVERSPEED OR UNDERSPEED. CONTINUE FLIGHT IN THE MANUAL THROTTLE CONTROL.

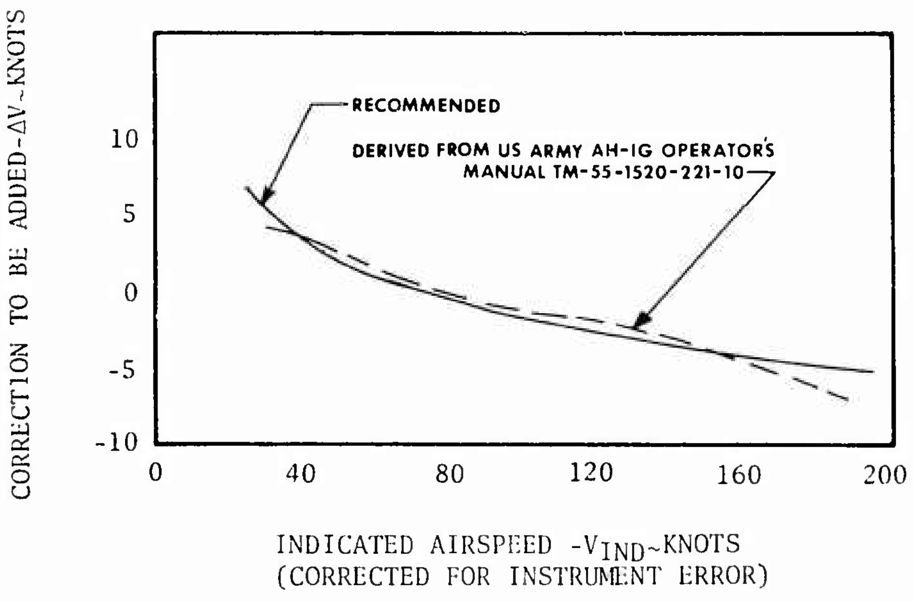
AIRSPEED CALIBRATION

80. Airspeed calibration tests were conducted to determine the position error of the standard and test (boom) airspeed systems in climb, dive, autorotation and level flight. The methods used to calibrate the test airspeed system were a combination of the trailing bomb, pacer aircraft and ground speed course test techniques. The calibration was conducted in the clean configuration only, and the data are presented in figures 131 and 132, appendix VII.

81. The standard airspeed system was calibrated using the trailing bomb and pacer aircraft methods, and the results are presented in figure 131, appendix VII. In addition to the data gathered during this evaluation, the test results include data from the AH-1G Phase B test reports (refs 2, 4 and 5, app I). In those test reports, the test configurations were clean, basic and outboard alternate. The position error in climb and autorotation was less than 3 knots from 55 to 100 KIAS and was acceptable. This airspeed band includes the airspeeds for maximum glide. Larger position errors were present from 30 to 55 KIAS, but these errors are not deemed significant since the helicopter is normally accelerating or decelerating through this airspeed band.

82. The standard airspeed system calibration for level and diving flight was compared to the position errors listed in the operator's manual (ref 19, app I). This comparison is presented in figure 11 and shows essentially the same position error from 40 to 170 KIAS. For the airspeed ranges from 30 to 40 KIAS and from 170 to 190 KIAS, there is a difference of 2 knots or less between the two sources of data. The airspeed position errors recorded during this test are satisfactory for the aircraft's mission and should be incorporated into the operator's manual.

FIGURE 11
AIRSPEED CALIBRATION
AH-1G T53-L-13
STANDARD AIRSPEED SYSTEM



CONCLUSIONS

GENERAL

83. Within the scope of this test, the AH-1G helicopter is suitable for the armed helicopter mission provided the insufficient directional control power and inadequate tail rotor drive system torque limitations are corrected (paras 17 and 19).

84. The AH-1G helicopter exceeded all contractor guarantees (paras 23, 32, 38, 40 and 42).

85. A directional control margin of 10 percent while hovering is the minimum acceptable for normal operation (para 17).

86. Installation of the engine inlet screens and the engine particle separator decrease the performance capability of the AH-1G when maximum power available is the limiting parameter (para 24).

87. The level-flight performance capabilities of the AH-1G vary with longitudinal cg location and improve as the cg moves aft (para 45).

88. The degradation of hover performance capability when hovering in adverse crosswind is significant (para 26).

89. The excellent climb performance, particularly from SL to 10,000 feet, enhances the capability of the AH-1G for the attack helicopter mission (para 31).

DEFICIENCIES AND SHORTCOMINGS AFFECTING MISSION ACCOMPLISHMENT

90. Correction of the following deficiencies is mandatory for successful accomplishment of the intended mission:

a. Insufficient directional control limits hovering, takeoff and landing performance (paras 16, 25 and 26).

b. The tail rotor drive system components are susceptible to damage due to the excessive tail rotor horsepower required for hovering flight (para 19).

91. Correction of the following shortcomings is desirable for improved operation and mission capability:

a. The inability to achieve maximum tail rotor blade angle (19 degrees) when full directional control is applied for all conditions with the present directional control/yaw SCAS geometry (para 20).

b. Moderate pilot effort required to maintain optimum climb airspeeds (para 34).

c. Extensive pilot compensation required to maintain maximum endurance airspeeds (para 44).

d. The possibility of inadvertently exceeding the main transmission torque limit due to the torque rise following a left-lateral control input when below the engine critical altitude (para 71).

RECOMMENDATIONS

92. The data presented in this report should be included in the operator's manual.

93. The deficiencies should be corrected on a high-priority basis.

94. The shortcomings should be corrected at the earliest convenience.

95. The operational flight envelope should be restricted to conditions which provide a 10-percent directional control margin (para 21a).

96. Initiate action to increase directional control margins and improve the torque transfer capability of the tail rotor drive system.

97. The following items should be included in the AH-1G operator's manual:

a. A warning to avoid hovering at 3- to 15-foot skid heights (para 21a).

b. A description of the modified level-flight acceleration takeoff technique (para 30).

c. An increase in the maximum climb airspeeds for night or instrument flight operations (para 34).

d. A discussion of the pilot's increased workload requirements when flying at maximum endurance airspeed (para 44).

e. The compatible engine torque limits (include throughout the manual) (para 66).

f. The revised procedure and warning notes for engine restart during flight (para 79).

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APPENDIX II. BASIC AIRCRAFT INFORMATION AND OPERATING LIMITS

AIRFRAME

Rotor System

The 540 "door hinge" main rotor assembly is a two-bladed, semi-rigid, underslung feathering axis type rotor. The assembly consists basically of two all-metal blades, blade grips, yoke extensions, yoke trunnion, and rotating controls. Control horns for cyclic and collective control input are mounted on the trailing edge of the blade grip. Trunnion bearings permit rotor flapping. The blade grip to yoke extension bearings permit cyclic and collective pitch action.

Tail Rotor

The tail rotor is a two-bladed, delta-hinge type employing preconing and underslinging. The blade and yoke assembly is mounted to the tail rotor shaft by means of a delta-hinge trunnion. Blade pitch angle is varied by movement of the tail rotor control pedals. Power to drive the tail rotor is supplied by a takeoff on the lower end of the main transmission.

Transmission System

The transmission is mounted forward of the engine and coupled to the engine by a short drive shaft. The transmission is basically a reduction gear box which transmits engine power at reduced rpm to the main and tail rotors by means of a two-stage planetary gear train. The transmission incorporates a free-wheeling clutch unit at the input drive. This provides a disconnect from the engine in case of a power failure to allow the aircraft to make an autorotational landing.

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.

Control Systems

A dual hydraulic control system is provided for the cyclic and collective controls. The directional controls are powered by a single servo cylinder which is operated by system number 1. The hydraulic system consists of two hydraulic pumps, two reservoirs, relief valves, shut-off valves, pressure warning lights, lines, fittings, and manual, dual tandem, servo actuators incorporating irreversible valves. Tandem power cylinders incorporating closed center four-way manual servo valves and irreversible valves are provided in the lateral, fore and aft cyclic and collective control system. A single power cylinder incorporating a closed center four-way manual servo valve is provided in the directional control system. The cylinders contain a straight-through mechanical linkage.

Force Trim

Magnetic brake and force gradient devices are incorporated in the cyclic control and directional pedal controls. These devices are installed in the flight control system between the cyclic stick and the hydraulic power cylinders and between the directional pedals and the hydraulic power cylinder. The force trim control can be turned off by depressing the left button on the top of the cyclic stick. The gradient is accomplished by springs and magnetic brake release assemblies which enable the pilot to trim the controls as desired.

Cyclic Control Stick

The pilot's and gunner's cyclic stick grips each have a force trim switch and a SCAS release switch. The pilot's cyclic stick has a built-in operating friction. The cyclic control movements are transmitted directly to the swash plate. The fore and aft cyclic control linkage is routed from the cyclic stick through the SCAS actuator, to the dual boost hydraulic actuater and then to the right horn of the fixed swash plate ring. The lateral cyclic is similarly routed to the left horn.

Collective Pitch Control

The collective pitch control is located to the left of the pilot and is used to control the vertical mode of flight. Operating friction can be induced into the control lever by hand tightening the friction adjuster. The pilot's and gunner's collective pitch controls have a rotating grip-type throttle.

Tail Rotor Pitch Control Pedals

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.

Stability and Control Augmentation System (SCAS)

The SCAS is a three-axis, limited-authority, rate-referenced stability augmentation system. It includes an electrical input which augments the pilot's mechanical control input. This system permits separate consideration of airframe displacements caused by external disturbances from displacements caused by pilot input. The SCAS is integrated into the fore, aft, lateral and directional flight controls to improve the stability and handling qualities of the helicopter. The system consists of electro-hydraulic servo actuators, control motion transducers, a sensor/amplifier unit and a control panel. The servo actuator movements are not felt by the pilot. The actuators are limited to a 25-percent authority and will center and lock in case of an electrical and/or a hydraulic failure.

ENGINE

Engine Description

The T53-L-13 engine, rated at 1400 shp, is a successor to the T53-L-11 engine. The additional power has been achieved with no change in the basic T53-L-11 engine envelope mounting and connection points and with a 6-percent increase in basic engine weight.

The performance gain is accomplished thermodynamically by the mechanical integration of a modified axial compressor, a two-stage compressor turbine and a two-stage power turbine into the T53-L-11 engine configuration.

Replacement of the first two compressor stators and changing of the first two stages of compressor rotor blades and disks results in an approximate 20-percent increase in mass air flow through the engine. This is accomplished without the use of inlet guide vanes.

An inlet flow fence, located on the outer wall of the inlet housing in the area of the previously used inlet guide vanes, provides the desired inlet conditions for the transonic compression during acceleration at low speeds. At compressor speeds up to 70 percent, the fence is in the extended position. Above 70 percent, the flow fence is retracted into the outer wall of the inlet housing. Similar to a piston ring, the circumference of the flow fence is changed by the action of a piston actuator powered by compressor discharge pressure.

The specification for this engine allows the use of JP-4 or JP-5 fuel for satisfactory operation throughout the engine's operating envelope. During this program, JP-4 fuel was used.

Engine Power Control System

The fuel control for the T53-L-13 engine is a hydro-mechanical type of fuel control. It 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.

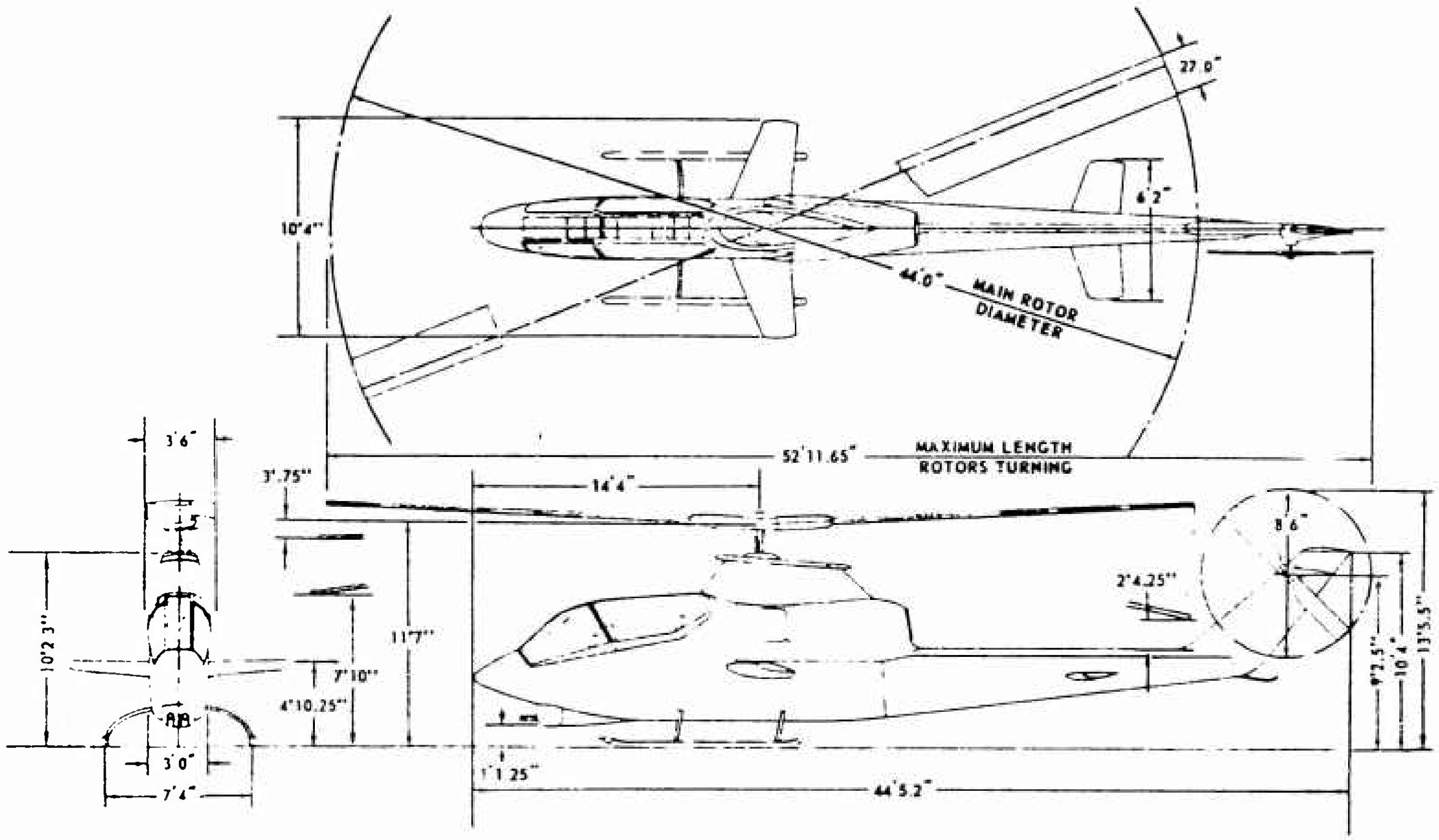
An air bleed control is incorporated within the fuel control to provide for opening and closing the compressor interstage air bleed in response to the following signals present in the power control:

a. Gas producer speed.

b. Compressor inlet air temperature.

c. Fuel flow.

The fuel control is designed to be operated either automatically or in an emergency mode. In the emergency position, fuel flow is terminated to the main metering valve and is routed 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 conected directly to the power control (twist grip). During the emergency operation, there is no automatic control of fuel flow during acceleration and deceleration; thus, EGT and engine acceleration must be pilot monitored.



Three-view drawing - AH-1G

BASIC AIRCRAFT INFORMATION

Airframe Data

Overall length (rotor turning)	637.2 inches
Overall width (rotor trailing)	124.0 inches
Center line of main rotor to center line of tail rotor	520.7 inches
Center line of main rotor to elevator hinge line	198.6 inches
Elevator area (total)	15.2 square feet
Elevator area (both panels)	10.9 square feet
Elevator airfoil section	Inverted Clark Y
Vertical stabilizer area	18.5 square feet
Vertical stabilizer airfoil section	Special camber
Vertical stabilizer aerodynamic center	FS 499.0
Wing area:	
Total	27.8 square feet
Outboard of BL 18.0 (both sides)	18.5 square feet
Wing span	10.33 feet
Wing airfoil section:	
Root	NACA 0030
Tip	NACA 0024
Wing angle of incidence	14 degrees

Main Rotor Data

Number of blades	2
Diameter	44 feet
Disc area	1520.5 square feet
Blade chord	27 inches
Rotor solidity	0.0651
Blade area (both blades)	99 square feet
Blade airfoil	9.33 percent symm special section
Linear blade twist	-0.455 deg/ft
Hub precone angle	2.75 degrees

Antitorque Rotor Data

Number of blades	2
Diameter	8.5 feet
Disc area	56.74 square feet
Blade chord	8.41 inches
Rotor solidity	0.105
Blade airfoil	NACA 0010 modified
Blade twist	Zero degrees

Transmission Drive System Ratios

Engine to main rotor	20.383:1.0
Engine to antitorque rotor	3.990:1.0
Engine to antitorque drive system	1.535:1.0

Test Aircraft Control Displacements

Longitudinal cyclic control:	
Full forward to full aft with SCAS nulled	9.07 inches
Lateral cyclic control:	
Full left to full right with SCAS nulled	10.00 inches
Directional (pedal) control:	
Full left to full right with SCAS nulled	7.07 inches
Collective control:	
Full up to full down with SCAS nulled	9.30 inches

OPERATING LIMITATIONS

Limit Airspeed (V_L)

Any configuration with XM159 rocket pods: 180 KCAS below a 3000-foot density altitude; decrease 8 KCAS per 1000 feet above 3000 feet

For this test, the AH-1G with skid gear fairings removed: same as standard configurations (Normal limit for operational use: 160 KCAS)

All other configurations: 190 KCAS below a 4000-foot density altitude; decrease 8 KCAS per 1000 feet above 4000 feet

Gross Weight/Center of Gravity Envelope

Forward center of gravity limit: Below 7000 pounds, FS 190.0; linear increase to FS 192.1 at 9500 pounds

Aft center of gravity limit: Below 8270 pounds, FS 201.0; linear decrease to FS 200 at 9500 pounds

Sideslip Limits

Five degrees at V_L with linear increase to 20 degrees at 60 KCAS

Rotor and Engine Speed Limits (Steady State)

Power on:	
Engine rpm	6400 to 6600
Rotor rpm	314 to 324
Power off:	
Rotor rpm	294 to 339
Rotor rpm transient lower limit	250
Power on during dives and maneuvers:	
Rotor rpm	314 to 324

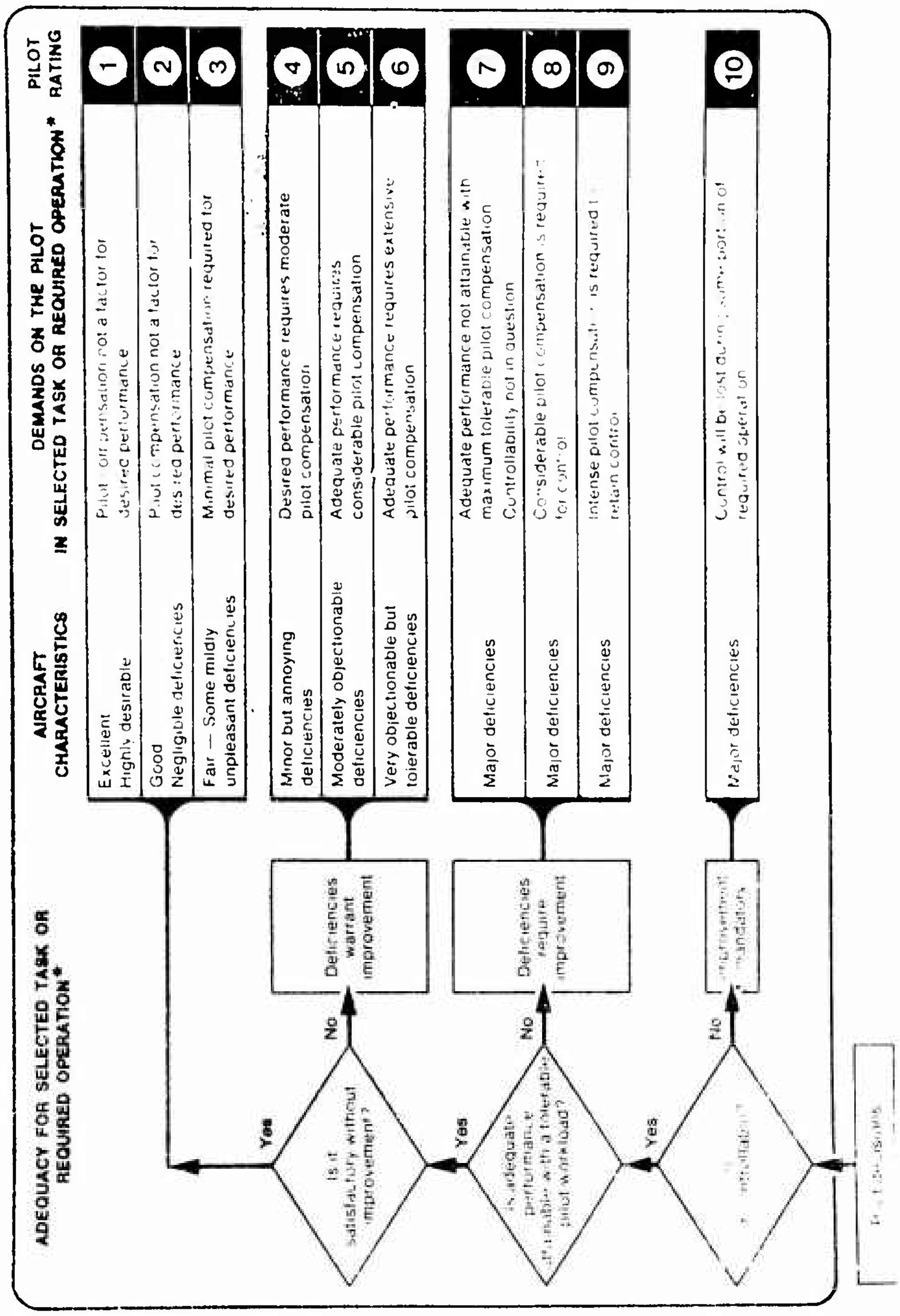
Temperature and Pressure Limits

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

T53-L-13 Engine Limits

Normal rated EGT (maximum continuous)	625°C
Military rated EGT (30-minute limit)	645°C
Starting and acceleration EGT (5-second limit)	675°C
Maximum EGT for starting and acceleration	760°C
Torque pressure limit	50 psi

APPENDIX III. HANDLING QUALITIES RATING SCALE



APPENDIX IV. 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 results obtained during this test program.

$$\text{Power Coefficient} = C_p = \frac{550\ \text{SHP}}{\rho A\ (\Omega R)^3}$$

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

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

$$\text{Main Rotor Tip Mach Number} = M_{tip} = \frac{1.689\ V_T + \Omega R}{a}$$

Instrumentation

2. All instrumentation was calibrated prior to commencing the test program. A detailed tabulation of the instrumentation is given in appendix V. All quantitative data obtained during this flight test program were derived from special sensitive instrumentation. Data were obtained from four aircraft sources and two ground support sources. The aircraft sources were: oscillograph, photopanel, pilot's panel (hand recorded) and engineer's panel (hand recorded). The ground support sources were: ground station and Fairchild camera station.

Weight and Balance

3. A high degree of control was maintained on weight and balance of the test helicopter. Variations in empty gross weight and cg

due to changes in helicopter component instrumentation were defined by periodically weighing the helicopter.

4. The empty weight of the test aircraft without instrumentation installed could not be determined since the aircraft was partially instrumented when it was delivered to USAAVNTA (USAASTA) at the beginning of the program. In addition, the aircraft was not a production model and was not representative of a standard AH-1G. The fuel load of the aircraft was defined by measuring the fuel specific gravity and temperature after each fueling, and by using an external sight gage on the calibrated fuel cell to determine fuel volume. Fuel used in flight was recorded by a calibrated fuel-used system, and the results were cross-checked with the sight gage reading following each flight. Helicopter loading and cg were controlled by ballast installed at various locations in the aircraft.

ANTITORQUE SYSTEM PERFORMANCE

5. The performance of the antitorque rotor system was defined by measuring various parameters. These parameters were recorded in hover, translation and forward flight. When the helicopter was stable, 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 and level flight conditions.

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

$$SHP_{TR} = TRQ_{TR} \times N_{TR} \times \frac{2\pi}{12 \times 33{,}000} \qquad (1)$$

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

$$C_{P_{TR}} = \frac{SHP_{TR} \times 550}{\rho A_{TR} (\Omega_{TR} R_{TR})^3} \qquad (2)$$

8. The tail rotor thrust for hover was determined by first making several assumptions. The three following assumptions were necessary since sufficient information about important parameters was not available to the test team: The first assumed that all restoring directional moment to maintain stabilized hover be attributed to 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. The second assumed that the total horsepower loss, attributed to frictional losses (gears, bearings, etc.) and horsepower extracted from main transmission to drive accessories (hydraulic pumps), was assumed to be 5 percent of the engine output shaft horsepower. This assumption was necessary to determine the horsepower delivered to the main rotor. The third assumption was necessary to determine the air density in the vicinity of the tail rotor. This analysis assumed that the free air temperature of the air mass flow passing through the tail rotor was not significantly influenced by the hot gases being emitted from the engine.

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

$$SHP_{MR} = SHP_{ENG} - SHP_{TR} - (0.05 \times SHP_{ENG}) \tag{3}$$

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

$$C_{P_{MR}} = \frac{SHP_{MR} \times 550}{\rho A \ (\Omega R)^3} \tag{4}$$

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

$$THRUST_{TR} = TRQ_{MR}/\ell_t = \frac{550 \ SHP_{MR}}{\Omega_{MR} \ \ell_t} \tag{5}$$

12. Equation 5 was expanded to obtain the nondimensional thrust coefficient of the tail rotor:

$$C_{T_{TR}} = \frac{C_{P_{MR}} \ R \ A \ (\Omega R)^2}{\ell_t \ A_{TR} \ (\Omega_{TR} \ R_{TR})^2} \tag{6}$$

13. The position of the directional control was determined by measuring pedal position with SCAS in the nulled position. Full left directional control application resulted in the tail rotor blade angle of 19 degrees for the test aircraft with SCAS in the nulled position. The total directional control (pedal) displacement (full left to full right) resulted in a 30.0-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 anti-torque data were obtained simultaneously with hover, translational and forward flight tests.

HOVER

15. To define hover performance, both the tethered and free-flight techniques were used. During tethered hovering, a helicopter cargo hook was secured to the bottom of the main transmission by a cable. An intermediate cable was then attached to a cable anchored to the ground. The length of this cable was varied to achieve the desired skid height. A load cell was installed between the helicopter and the ground to measure cable tension. Increasing cable tension had the same effect on hovering performance as increasing gross weight. When power required and cable tension were stabilized, the parameters necessary to define gross weight, cable tension shaft horsepower and ambient air conditions were recorded. During free-flight hovering tests, the helicopter was stabilized at a skid height of 100 feet (OGE). When the helicopter was stable, the parameters to define gross weight, shaft horsepower and ambient air conditions were recorded. The free-flight hovering technique was used only at a skid height of 100 feet to provide a cross-check of tethered hovering technique. The clean configuration was used to gather a majority of the hovering data. A limited amount of hover data were gathered in the heavy hog configuration to determine the effects of wing stores armament on hovering data. All hovering performance tests were conducted in winds of less than 2 knots.

16. Hovering data collected in terms of gross weight, shaft horsepower and ambient air conditions were converted to define the relationship between the nondimensional 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.

17. The wind limitation envelope during hover and translational flight was determined by conducting tests at various combinations of azimuth and airspeed. When the aircraft was reasonably stabilized in translational flight, parameters necessary to determine gross weight, ambient air conditions, azimuth, airspeed and directional control (pedal) with SCAS in the nulled position were recorded. A ground vehicle with a calibrated speedometer was used as a pacer to determine true airspeed for each stabilized condition. Ambient wind velocity and direction were incorporated into

the analysis when determining the exact speed and direction of the aircraft when translating across the ground. Tests were conducted when wind velocities were less than 4 knots. The results of each individual test are presented in reference 15, appendix I, and are summarized in this report in nondimensional and engineering unit forms.

TAKEOFF

18. Takeoff performance was defined by measuring the horizontal distance required to takeoff and clear an obstacle 50 feet high as a function of airspeed. This distance was primarily a function of airspeed and the magnitude of engine power available above that required to hover at a reference skid height. The reference skid height used during this test program was 3 feet. This takeoff performance, expressed in nondimensional terms, is shown in the following equation:

$$\Delta C_P = C_P \text{ available at test conditions} \qquad (7)$$

$$- C_P \text{ required to hover at a 3-foot skid height}$$

19. A series of takeoffs was conducted at a single ΔC_P throughout an airspeed of range. This series defined the variation in takeoff distance versus airspeed for a single ΔC_P. Day-to-day temperature variation permitted testing through a range of ΔC_P by changing only gross weight and pressure altitude. Curves of distance required to clear a 50-foot obstacle versus airspeed at various values of ΔC_P were carpet-plotted. This carpet-plot defined takeoff performance throughout a wide range of gross weights, pressure altitudes, ambient temperatures and airspeeds. All tests were conducted with winds of less than 4 knots. A Fairchild flight analyzer was used to determine horizontal and vertical distances and true airspeeds.

CLIMB

20. Continuous-climb performance tests were conducted by establishing engine power (1100 shp) at a transmission input torque limit below critical engine altitude and military power above the critical altitude. The airspeed schedule used during all climb tests was derived from the level-flight performance data. All climbs were flown at an airspeed which produced the maximum engine power differential between engine power required for level flight and engine power available. All climbs except the climb for contract guarantee

compliance check were flown from near SL to service ceiling. The climb to check contract guarantee compliance was flown from near SL to an approximate 9000-foot H_D. Climbs were conducted at two gross weights in both the clean and heavy hog configuration. Additional climbs were flown in these two configurations from near SL to a 10,000-foot H_D to determine the climb power coefficient (K_p) and the climb gross weight coefficient (K_w).

21. Climb tests were conducted on nonstandard days; therefore, several corrections were necessary to define standard day climb performance. The observed rate of change in pressure altitude was converted to tapeline rate of climb by the expression:

$$R/C_{tapeline} = dhp/dt \; (T_t/T_{std}) \tag{8}$$

22. At the test density altitude, the variation in rate of climb for nonstandard power available was calculated by the expression:

$$\Delta R/C_{power} = K_p \frac{(SHP_{std} - SHP_t)\;(33,000)}{GRWT_t} \tag{9}$$

23. The variation in rate of climb for nonstandard gross weight was calculated by the expression:

$$\Delta R/C_{weight} = K_w \frac{SHP_s \times 33,000\;(GRWT_t - GRWT_s)}{GRWT_s\;GRWT_t} \tag{10}$$

24. The standard day rate of climb was then calculated:

$$R/C_{std} = R/C_t + \Delta R/C_{power} + \Delta R/C_{weight} \tag{11}$$

LEVEL FLIGHT

25. Level flight performance was defined by measuring the shaft horsepower required to maintain level flight throughout the airspeed range of the helicopter. A constant C_T was maintained by increasing altitude as fuel was consumed. A broad range of C_T's was flown for eight different wing store configurations at a forward cg and with the landing gear cross-tube fairings removed. The results of the level-flight tests were converted to nondimensional form and carpet-plotted as C_P versus C_T with lines of constant tip-speed ratio. This carpet-plot defined the level flight performance for all gross weights, density altitudes and airspeeds throughout

the range of C_T's tested for each aircraft configuration.

26. Specific range performance was calculated from the relationship of the true airspeed at any power setting to the engine fuel flow at that power setting. For any given gross weight and standard day ambient conditions, the following would apply:

$$\text{Specific Range} = \frac{\text{true airspeed}}{\text{fuel flow}} = \frac{\text{nautical air miles}}{\text{per pound of fuel}} \quad (12)$$

27. Fuel flow at any power setting and standard day atmospheric conditions was derived from engine model specification 104.33 for the T53-L-13 engine (ref 18, app I). All faired, level-flight information based on fuel-flow data from reference 18, appendix I, include 5-percent conservatism per MIL-C-5011A (ref 21).

28. Increase in equivalent flat plate area for various wing store and aircraft configurations was calculated by the following equation:

$$\Delta f = \frac{2\ \Delta C_P\ A\ (\Omega R)^3}{(V_T \times 1.689)^3} = \frac{2\ \Delta C_P\ A}{\mu^3} \quad (13)$$

29. This method for evaluating equivalent flat plate area was valid only for airspeeds above 90 KTAS.

Autorotation

30. Autorotational descent performance data were acquired during sawtooth autorotations. Variation in rate of descent with airspeed was defined by stabilizing at a constant airspeed with a rotor speed of 324 rpm and measuring rate of descent. To determine the effect of rotor speed on rate of descent, airspeed was stabilized and rotor speed was varied. The observed rate of descent was corrected to tapeline rate of descent by the expression:

$$R/D_{tapeline} = (dhp/dt)(T_t/T_{std}) \quad (14)$$

Power Determination

31. The engine torquemeter is essentially a piston (restrained by oil); the pressure of which is proportional to the power output of the engine. The equation for determining the test shp as obtained from engine manufacturer test cell calibration curves is developed as outlined in paragraphs 32 through 36.

32. The horsepower transmitted by a rotating shaft may be expressed in the following manner:

$$SHP = \frac{2\pi}{12 \times 33{,}000} \times N_E \times TRQ \qquad (15)$$

33. The calibration of the engine's torquemeter system for engine S/N LE14001 indicated that engine shaft output torque was slightly nonlinear as a function of indicated torque pressure. This nonlinear relationship for engine S/N LE14001 is graphically presented in figure I. The calibration range for engine S/N LE14008 was not sufficient to provide a valid means of determining engine output torque as a function of engine output torque pressure since the entire operating range was not covered. However, the limited amount of information available on this engine's torque measuring system is presented in figure II. These plots were used to obtain engine output torque.

34. The rotor speed can be determined from engine output shaft speed as follows:

$$N_R = \frac{N_E}{20.383} \qquad (16)$$

35. Substituting equation 16 into equation 15, a convenient equation for determining output shaft horsepower can be developed:

$$SHP = \frac{2\pi \times 20.383 \times TRQ \times N_R}{12 \times 33{,}000} = 3.234 \times 10^{-4} \times TRQ \times N_R \qquad (17)$$

36. This equation was used during the program to determine the shaft horsepower for each test condition.

ENGINE CHARACTERISTICS

Engine "Beep" Control Characteristics

37. The engine "beep" control characteristics were defined both with a loaded and unloaded main rotor system. The engine "beep" control characteristics were defined by stabilizing at a rotor speed of 324 rpm while in level flight and on the ground. The engine "beep" control was then actuated for a specified time. A continuous record was made of engine and rotor speed response during the maneuver. This process was repeated until the entire speed-range authority of the "beep" control was determined.

FIGURE No. I

ENGINE CHARACTERISTICS

T53-L-13 S/N LE14001

NOTES: 1. DASHED LINE OBTAINED FROM LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION No. 104.35

2. POINTS ENCLOSED WITH CIRCLES (○) OBTAINED FROM ENGINE MANUFACTURE'S CALIBRATION TEST CONDUCTED ON 22 AUGUST 1967.

3. POINTS ENCLOSED WITH SQUARES (□) OBTAINED FROM ENGINE MANUFACTURE'S CALIBRATION TEST CONDUCTED ON 7 APRIL 1969.

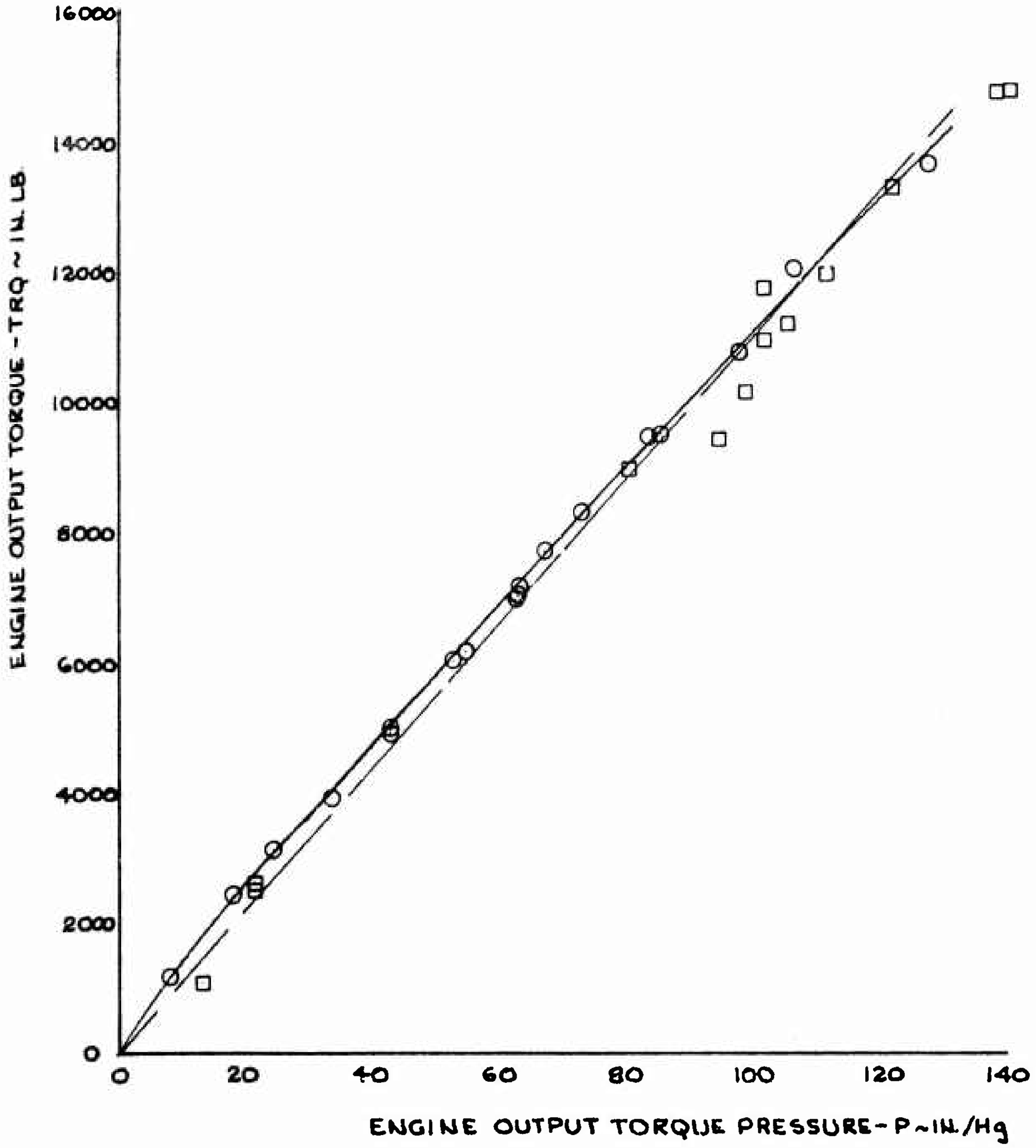
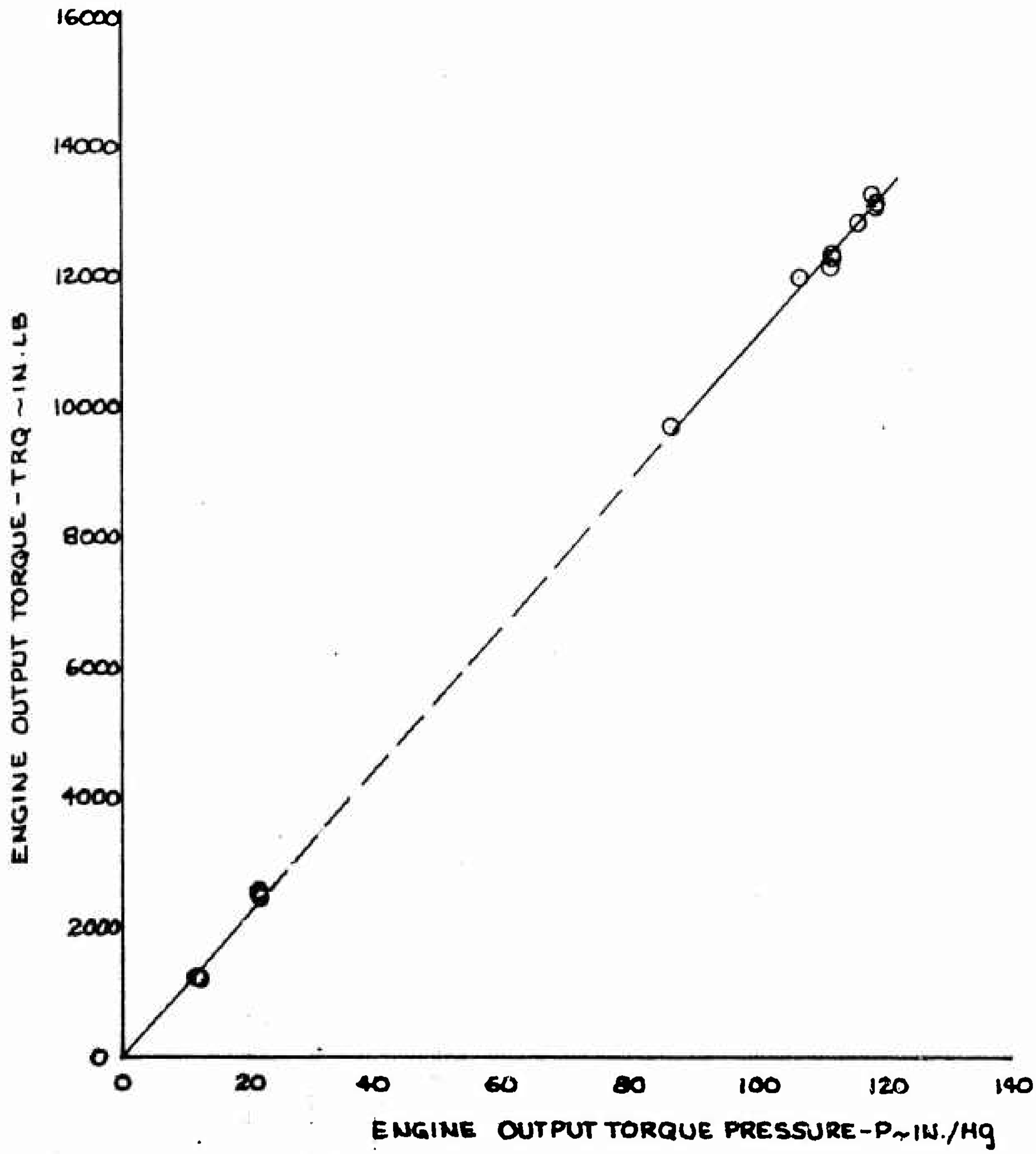


FIGURE No. II

ENGINE CHARACTERISTICS

T53-L-13 S/N LE14008

NOTE : POINTS OBTAINED FROM ENGINE MANUFACTURE'S CALIBRATION TEST CONDUCTED 28 AUG 1966



38. Test shaft horsepower and measured values of fuel flow, gas producer speed and exhaust gas temperature were corrected to standard day, SL atmospheric conditions. The engine characteristics are defined by the following equations:

$$\frac{N_1}{\sqrt{\theta_{t_2}}} \text{ versus } \frac{SHP}{\delta_{t_2}\sqrt{\theta_{t_2}}} \tag{18}$$

$$\frac{W_f}{\delta_{t_2}\sqrt{\theta_{t_2}}} \text{ versus } \frac{SHP}{\delta_{t_2}\sqrt{\theta_{t_2}}} \tag{19}$$

$$SFC \text{ versus } \frac{SHP}{\delta_{t_2}\sqrt{\theta_{t_2}}} \tag{20}$$

$$\frac{EGT}{\theta_{t_2}} \text{ versus } \frac{SHP}{\delta_{t_2}\sqrt{\theta_{t_2}}} \tag{21}$$

$$\frac{W_f}{\delta_{t_2}\sqrt{\theta_{t_2}}} \text{ versus } \frac{N_1}{\sqrt{\theta_{t_2}}} \tag{22}$$

Airspeed Calibration

39. The test airspeed indicator system (boom) and standard airspeed system were calibrated by comparing readings to a known reference. A calibrated trailing bomb was suspended from the helicopter with a cable approximately 50 feet in length to avoid proximity effect. The aircraft was then stabilized at various airspeeds in level flight, climb and autorotation. By comparing the airspeed corrected for instrument errors of both systems to the bomb system, the error was defined.

40. The test boom airspeed indicator system was calibrated at higher airspeeds, both in level flight and dive using a T-28 pacer aircraft. The test and pacer aircraft were stabilized at the same

airspeed, and data were recorded in each aircraft simultaneously. The calibrated airspeed was computed from the known position error of the pacer aircraft.

41. The test boom airspeed indicator system was calibrated in level flight over a measured ground course. Two passes were flown on reciprocal headings at each airspeed to average wind effects. This method provided a cross-check on the trailing bomb method described in paragraph 39.

42. The test boom airspeed system consisted of a boom with a non-swiveling pitot-static head mounted just aft and below the nose of the aircraft. This pitot-static system was connected to the sensitive airspeed and altimeter indicators on the instrument panels. This system was used in place of the standard pitot-static system since the standard system was not accurate when both systems were installed on the aircraft.

APPENDIX V. TEST INSTRUMENTATION

Flight test instrumentation was installed in the test helicopter prior to the start of this evaluation. This instrumentation provided data from four sources: pilot's panel, copilot/gunner's panel, photopanel, and a 24-channel oscillograph (see photos). All instrumentation was calibrated. The flight test instrumentation was installed and maintained by USAASTA. The following test parameters were presented.

PILOT'S PANEL

Standard system airspeed
Boom system airspeed
Boom system altitude
Rate of climb
Gas producer speed
Torque pressure (standard system)
Exhaust gas temperature
Longitudinal control position
Lateral control position
Pedal control position
Collective control position
Center of gravity (normal acceleration)
Angle of sideslip

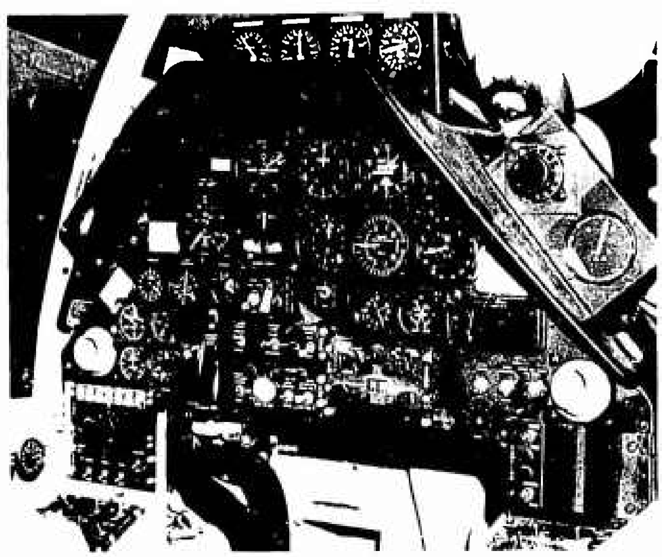


Photo 1. Pilot's Panel.

ENGINEER PANEL

Boom system airspeed
Boom system altitude
Outside air temperature
Rotor speed
Gas producer speed
Fuel used (total)
Torque pressure (high)
Torque pressure (low)
Exhaust gas temperature
Oscillograph correlation counter
Photopanel correlation counter
Fuel temperature
Engine fuel flow

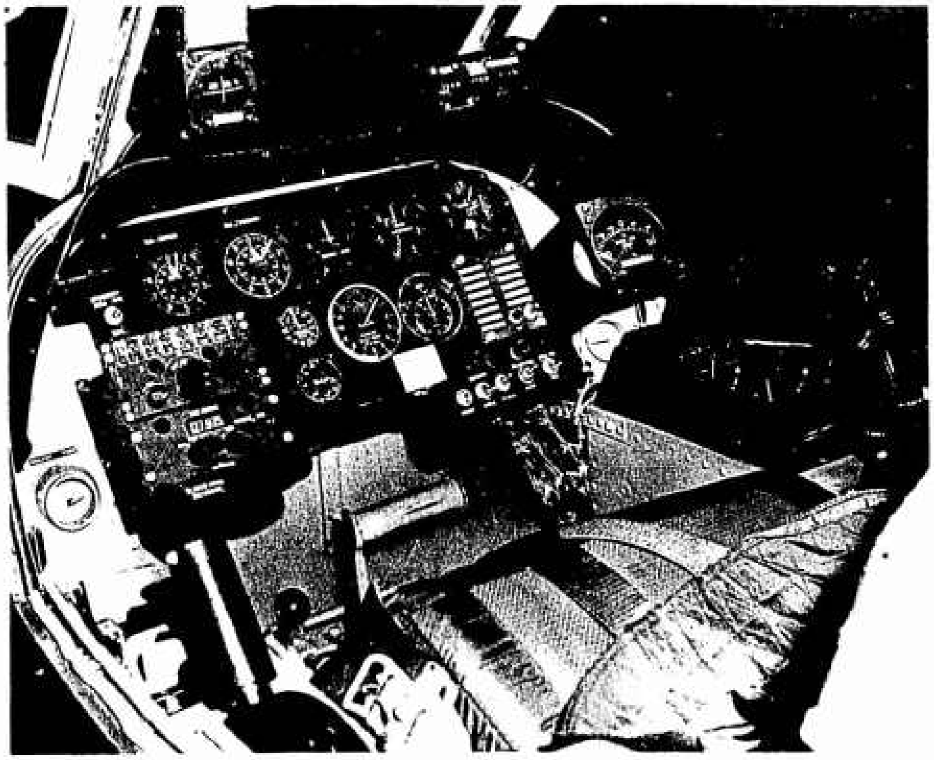


Photo 2. Copilot/Engineer's Panel.

PHOTOPANEL

Boom system airspeed
Standard system airspeed
Boom system altimeter
Rotor speed
Gas producer speed
Fuel used total
Torque pressure (high)
Torque pressure (low)
Exhaust gas temperature
Compressor inlet temperature
Compressor inlet total pressure
Inlet guide vane position
Bleed band position (light)
Fuel pressure at nozzle
Time (10-second stopwatch)
Oscillograph correlation counter
Photopanel correlation counter
Engineer's event
Pilot's event

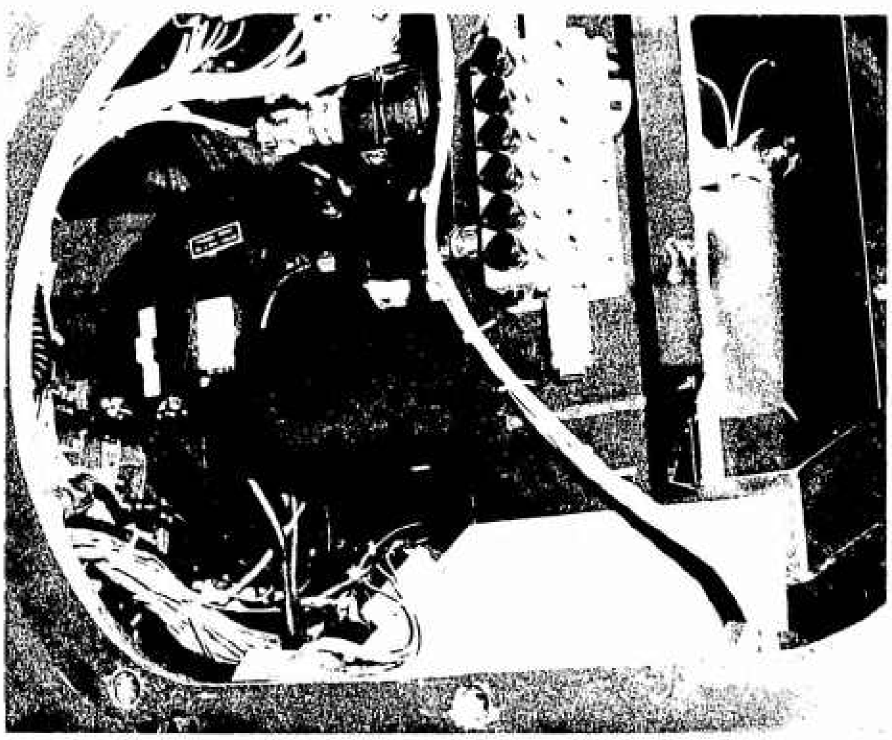


Photo 3. Photopanel.

OSCILLOGRAPH

Longitudinal control position
Lateral control position
Directional control position
Collective control position
Pitch attitude
Roll attitude
Yaw attitude
Pitch rate
Roll rate
Yaw rate
CG (normal acceleration)
Angle of sideslip
Angle of attack
Engineer's event
Pilot's event
Photopanel correlation blip
Linear rotor speed
Gas producer speed
Inlet guide vane position
Bleed band position
Fuel pressure at the nozzle
Tail rotor torque

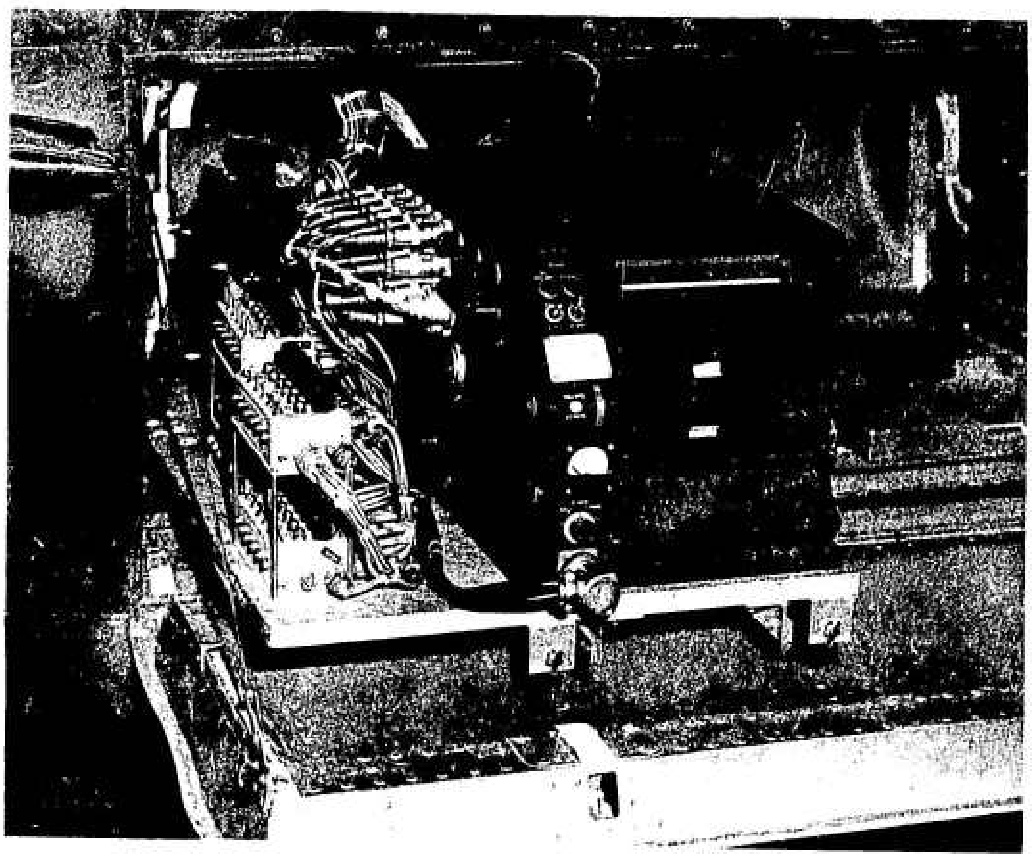


Photo 4. 24-Channel Oscillograph.

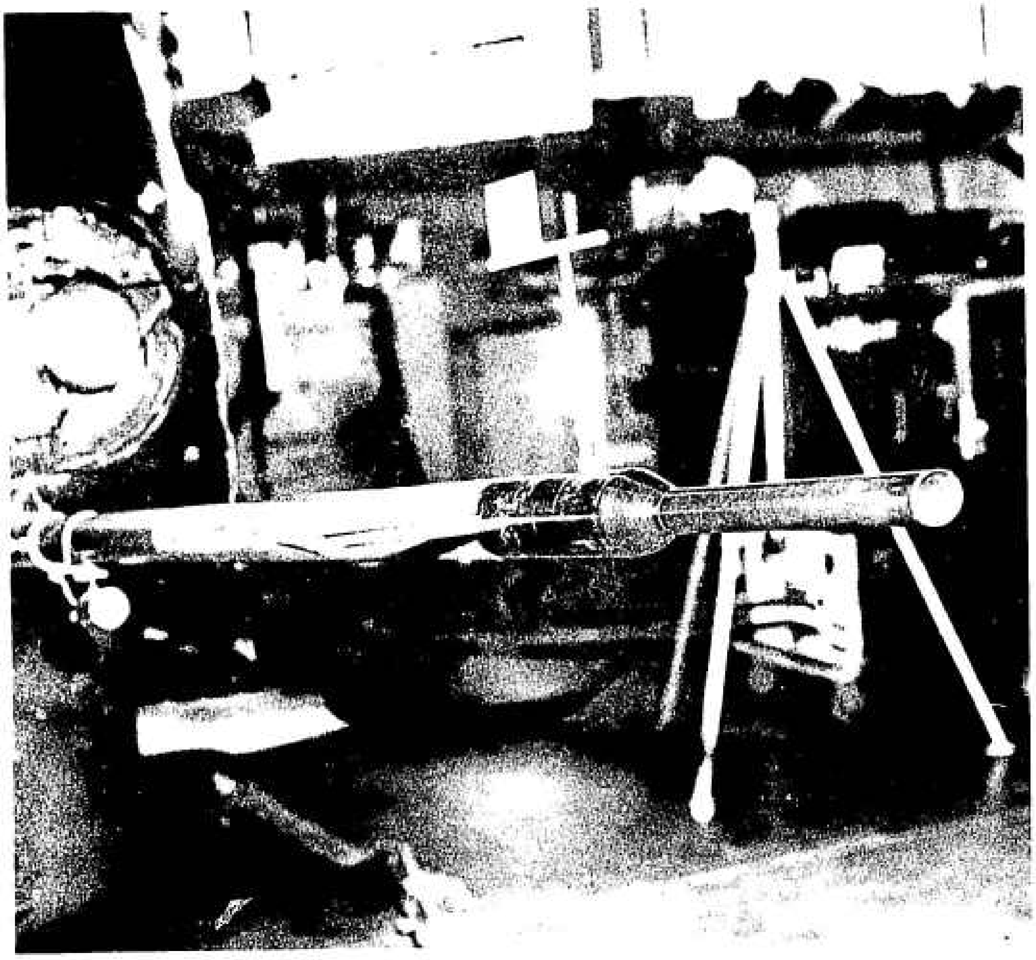


Photo 5. Test Pitot-Static System.

APPENDIX VI. CONTRACT GUARANTEES

1. A summary of the AH-1G helicopter contract guarantees and the results of the tests to determine compliance with these guarantees is shown in table A. The calculations to determine operating radius and endurance guarantees are included in tables B and C.

2. The aircraft shall be capable of the following performance under International Civil Aviation Organization (ICAO) standard air conditions (unless otherwise specified) at a gross weight of 8000 pounds. The installed armament shall be the XM28 turret and two LAU-3/A (XM159) 19 round rocket pods. Performance is predicated on the XM28 turret having the same aerodynamic drag as the TAT-102A turret. Engine fuel flow is based on engine Model Specification No. 104.33, for the Shaft Turbine Engine, Model T53-L-13, Lycoming Division of Avco Corporation, 30 September 1964, revised 30 July 1965 and 6 May 1966 using JP-4 fuel. All performance items were determined without the government furnished aircraft equipment (GFAE) particle separator or foreign object damage screen installed.

Table A. AH-1G Performance Guarantees

Performance Conditions	Units	Guaranteed	Test Results	
			Forward CG	Aft CG
Speed at SL (6600 rpm) (1100 shp).	Knots	144.0	140.0	153.0
Maximum endurance at SL with 1600 pounds of fuel. Fuel includes a 10-percent reserve plus warm-up and takeoff allowance. Does not include a 5-percent increase in engine specification sfc (6600 rpm).	Hours	3.0	3.03	3.08
Operating radius at cruising speed at SL with 1600 pounds of fuel. Fuel includes a 10-percent reserve plus warm-up and takeoff allowance. Does not include a 5-percent increase in engine specification sfc (6600 rpm).	NM	148.0	149.6	166.0
Best R/C at 1100 shp limit at SL (6600 rpm).	fpm	1800	1835	1900
Hover ceiling OGE (6600 rpm) with 95°F OAT (MRP).	Feet	2000	3390	3390
Vertical R/C 1100 shp limit at SL (6600 rpm).	fpm	500	Not Tested	Not Tested

Table B. Range Performance Contract Guarantee Analysis.

Configuration: outboard alternate Standard day Altitude: sea level	Rotor speed: 324 rpm Rocket pod fairings removed	
Condition	Aircraft Gross Weight (lb)	Fuel (lb)
Engine start conditions	8000	1600
Initial condition after fuel required for warm-up and takeoff has been consumed (assumed to be 25 pounds)	7975	1575
Final condition with a 10-percent fuel reserve	6560	160

FORWARD CG

Engine fuel flow values do not include a 5-percent increase in engine specification fuel flow.

Engine fuel flow for initial condition at maximum NAMPP: 598.5 lb/hr

Cruise airspeed for initial condition at maximum NAMPP: 124.5 NM/hr

Engine fuel flow for final condition at maximum NAMPP: 583.5 lb/hr

Cruise airspeed for final condition at maximum NAMPP: 125.5 NM/hr

Average fuel flow: $\frac{598.5 + 583.5}{2} = 591$ lb/hr

Average cruise airspeed: $\frac{124.5 + 125.5}{2} = 125$ NM/hr

Usable fuel: 1415 pounds

Distance traveled: 1415 lb x $\frac{1}{591 \text{ lb/hr}}$ x 125 NM/hr = 299.3 NM

Operating radius: $\frac{299.3}{2}$ = 149.6 NM

AFT CG

Fuel-flow values do not include a 5-percent increase in engine specification fuel flow.

Engine fuel flow for initial condition at maximum NAMPP: 588 lb/hr

Cruise airspeed for initial condition at maximum NAMPP: 135 NM/hr

Engine fuel flow for final condition at maximum NAMPP: 567 lb/hr

Cruise airspeed for final condition at maximum NAMPP: 136 NM/hr

Average fuel flow: $\frac{588 \text{ x } 567}{2}$ = 577.5 lb/hr

Average cruise airspeed: $\frac{135 \text{ x } 136}{2}$ = 135.5 NM/hr

Usable fuel: 1415 pounds

Distance traveled: 1415 lb x $\frac{1}{577.5 \text{ lb/hr}}$ x 135.5 NM/hr = 332.0 NM

Operating radius: $\frac{332.0}{2}$ = 166.0 NM

Table C. Endurance Performance Contract Guarantee Analysis.

Configuration: outboard alternate Standard day Altitude: sea level	Rotor speed: 324 rpm Rocket pod fairings removed	
Condition	Aircraft Gross Weight (lb)	Fuel Load (lb)
Engine start condition	8000	1600
Initial condition after fuel required for takeoff has been consumed (assumed to be 25 pounds)	7975	1575
Final condition with a 10-percent fuel reserve	6560	160

FORWARD CG

Engine fuel flow valves do not include a 5-percent increase in engine specification fuel flow.

Engine fuel flow for initial condition at minimum shp: 478.5 lb/hr

Engine fuel flow for final condition at minimum shp: 456 lb/hr

$$\text{Average fuel flow: } \frac{478.5 + 456}{2} = 467.3 \text{ lb/hr}$$

Usable fuel: 1415 pounds

$$\text{Endurance time: } \frac{1415 \text{ lb}}{467.3 \text{ lb/hr}} = 3.03 \text{ hr}$$

AFT CG

Engine fuel flow valves do not include a 5-percent increase in engine specification fuel flow.

Engine fuel flow for initial condition at minimum shp: 468 lb/hr

Engine fuel flow for final condition at minimum shp: 450 lb/hr

Average fuel flow: $\frac{468 + 450}{2}$ = 459 lb/hr

Usable fuel: 1415 lb

Endurance time: $\frac{1415 \text{ lb}}{459}$ = 3.08 hr

APPENDIX VII. TEST DATA

FIGURE NO. 1

DIRECTIONAL CONTROL MARGINS
AS A FUNCTION OF AIRCRAFT SKID HEIGHT
IN A HOVER

AH-1G USA S/N 615247

NOTES: 1. TOTAL DIRECTIONAL CONTROL DISPLACEMENT IS 7.07 INCHES
2. FULL LEFT DIRECTIONAL CONTROL = 19° TAIL ROTOR PITCH
3. WIND LESS THAN 2 KNOTS

CURVES DERIVED FROM FIGURES 2 THRU 7 APP VII

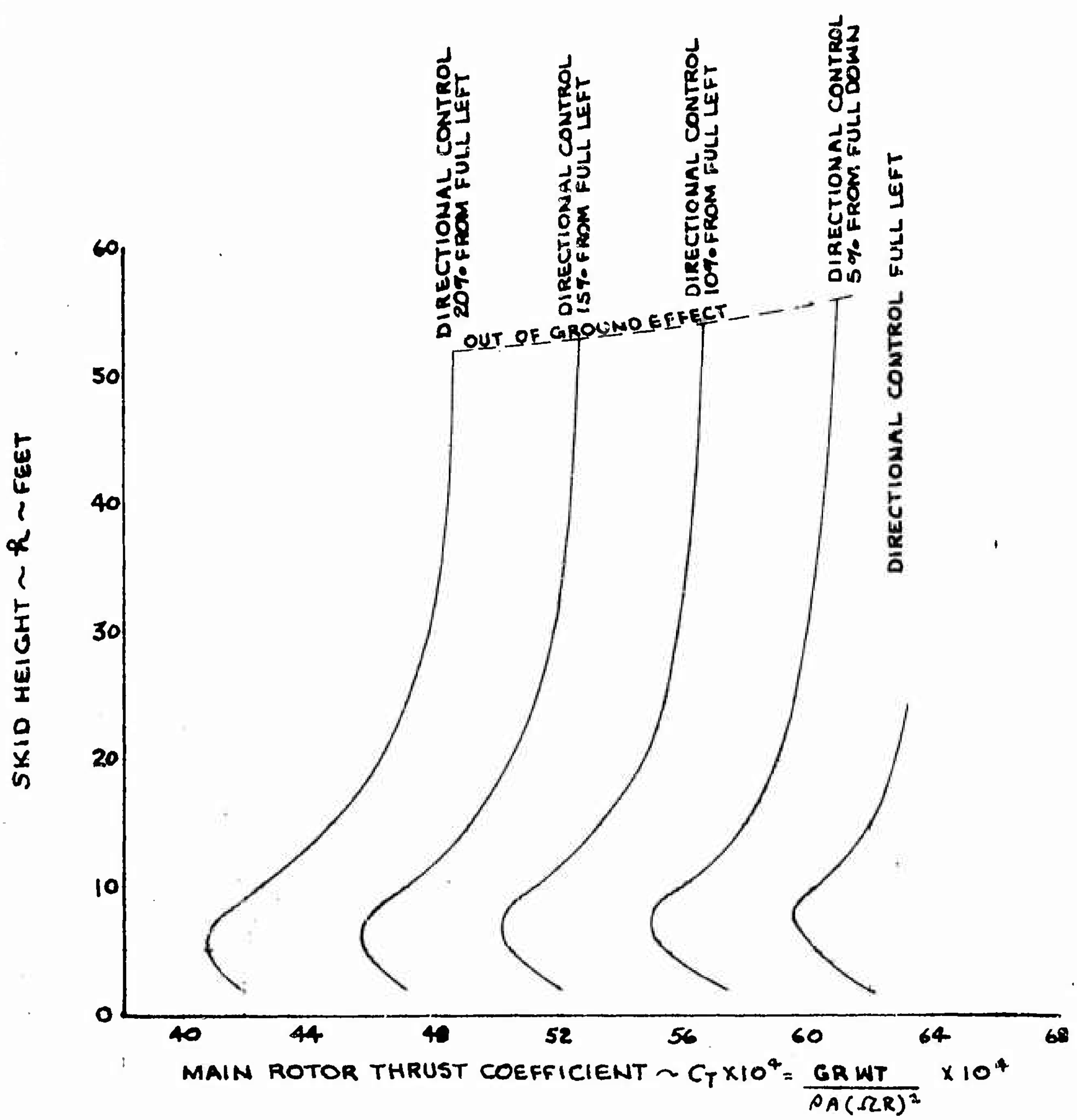


FIGURE NO. 2

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	ROTOR SPEED ~ RPM
○	324
□	314

NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. OPEN SYMBOLS DENOTE CLEAN CONFIGURATION
3. SOLID SYMBOLS DENOTE HOG CONFIGURATION
4. WIND LESS THAN 2 KNOTS
5. TOTAL DIRECTIONAL CONTROL DISPLACEMENT IS 7.07 INCHES FROM FULL LEFT
6. SKID HEIGHT = 2 FEET

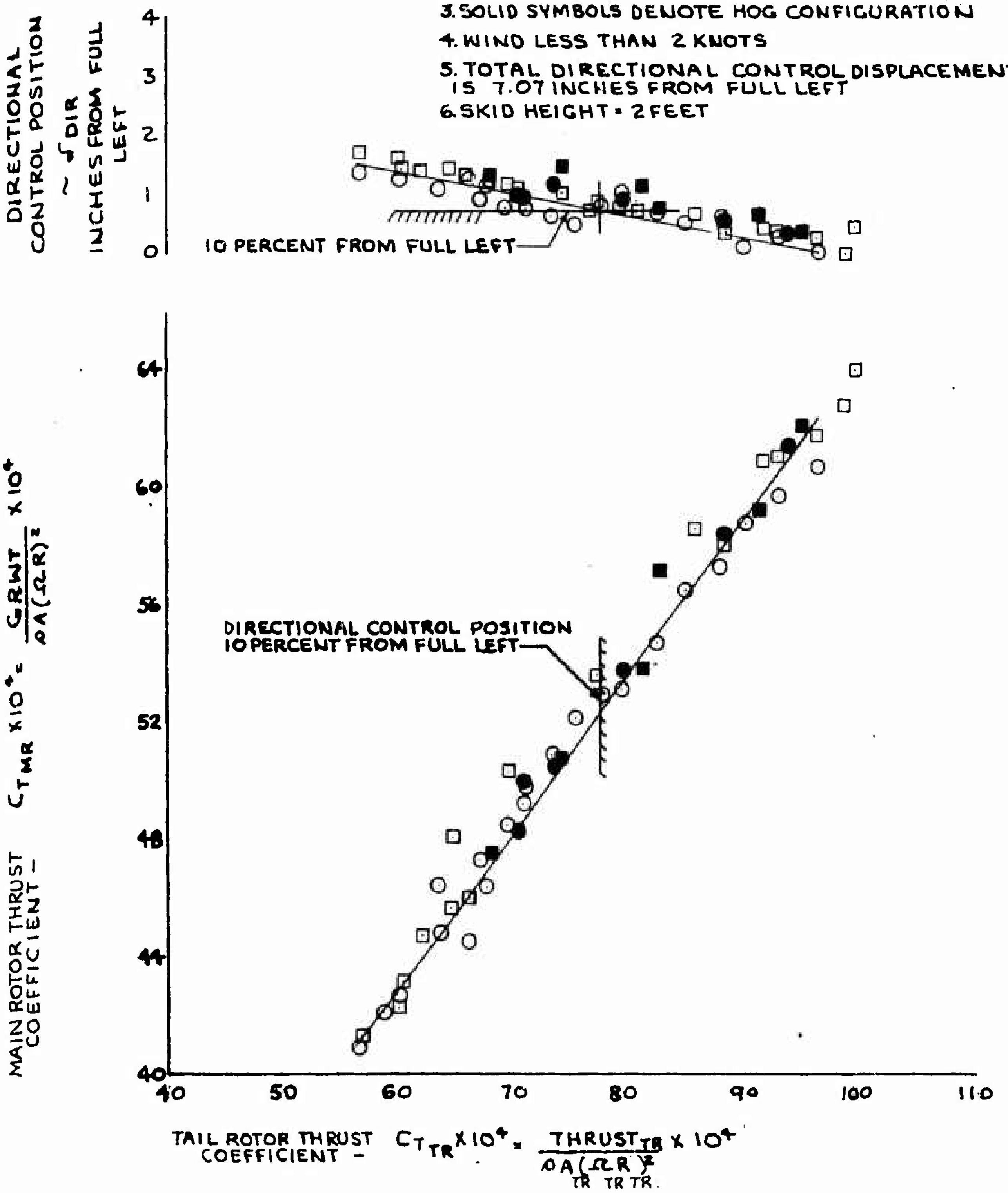


FIGURE No. 3

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

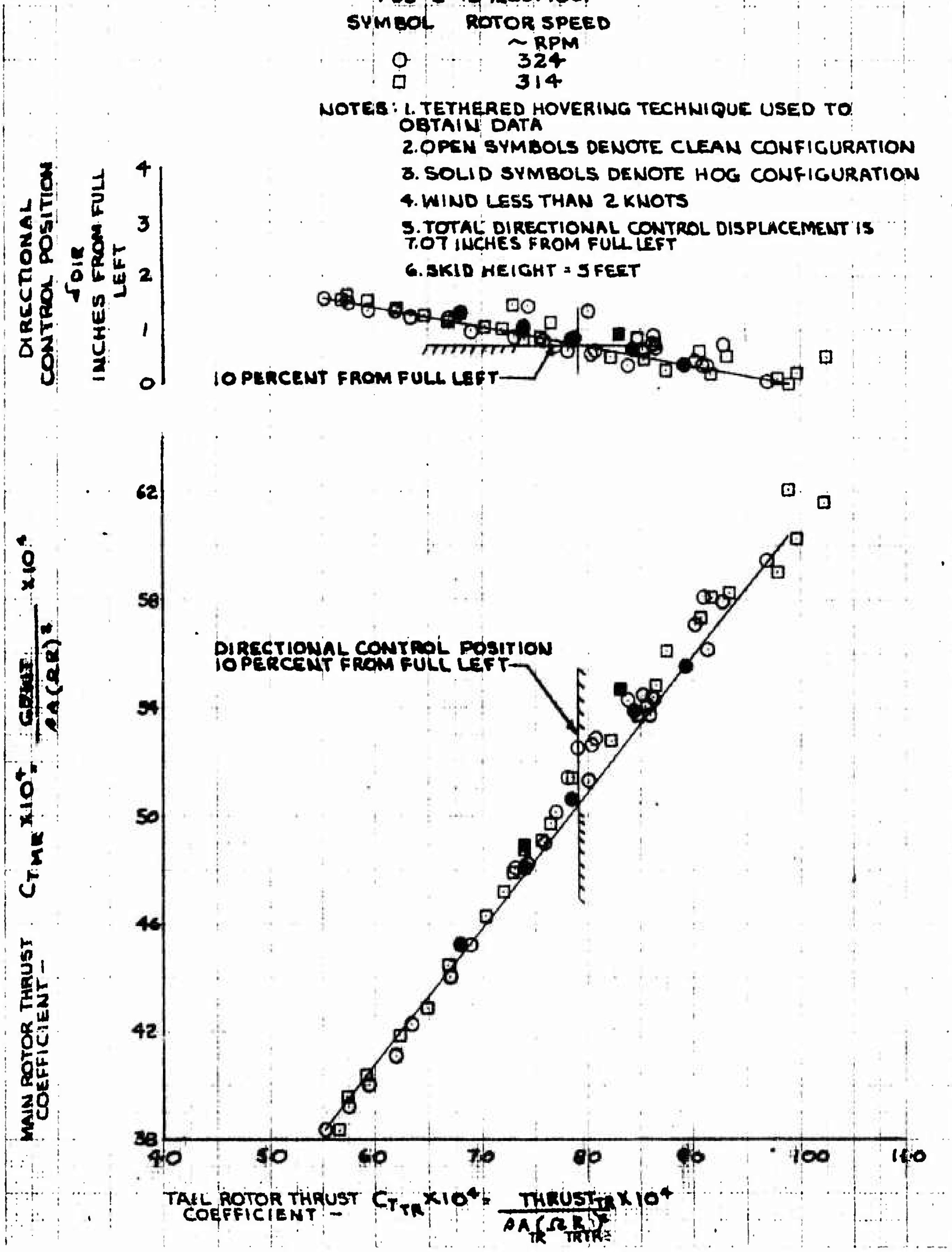


FIGURE No 4

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	ROTOR SPEED ~RPM
○	324
□	314

NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA

2. OPEN SYMBOLS DENOTE CLEAN CONFIGURATION
3. SOLID SYMBOLS DENOTE HOG CONFIGURATION
4. WIND LESS THAN 2 KNOTS
5. TOTAL DIRECTIONAL CONTROL DISPLACEMENT IS 7.07 INCHES FROM FULL LEFT
6. SKID HEIGHT = 10 FEET

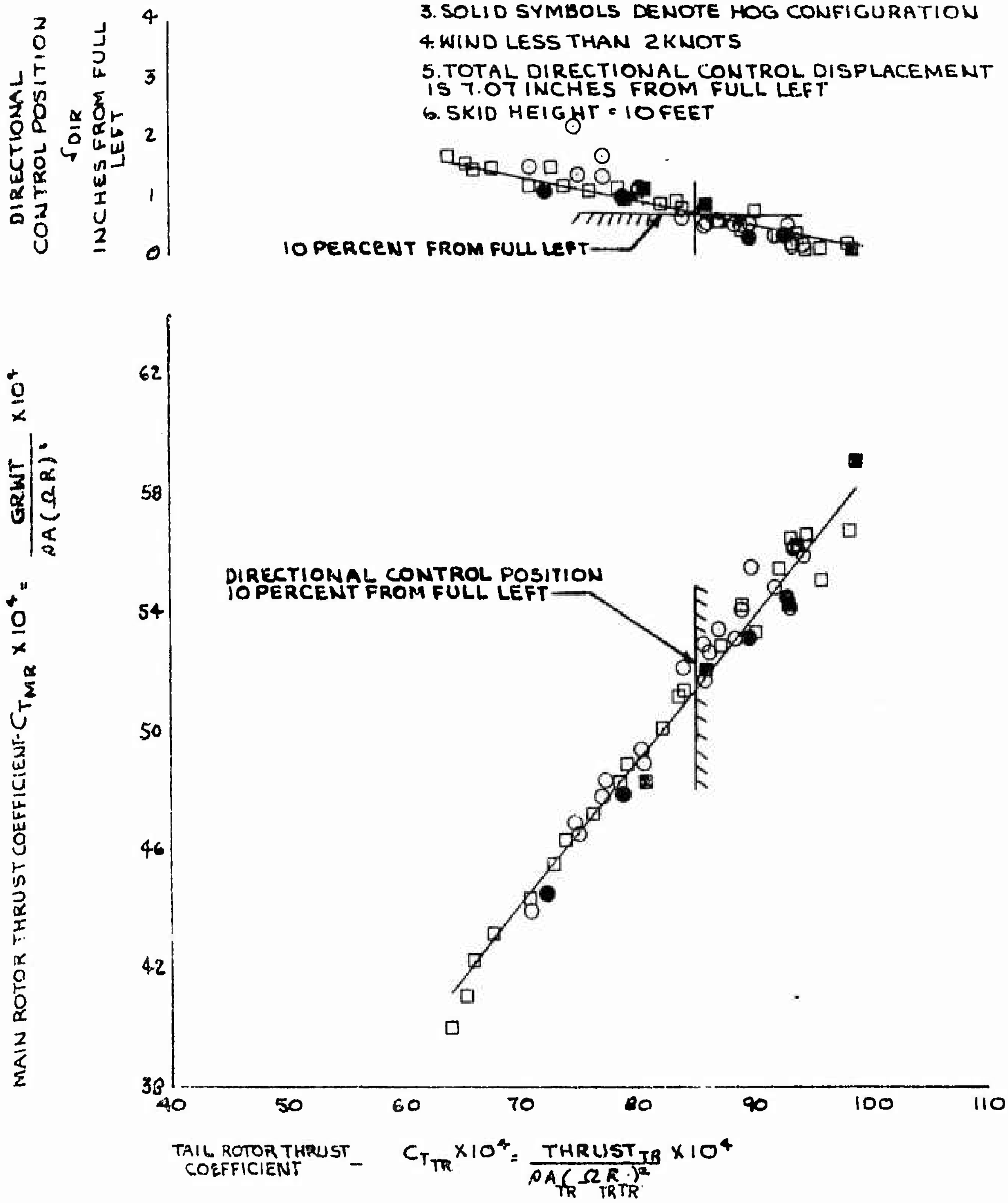


FIGURE No. 5

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	ROTOR SPEED ~ RPM
○	324
□	314

NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA

2. OPEN SYMBOLS DENOTE CLEAN CONFIGURATION

3. SOLID SYMBOLS DENOTE HOG CONFIGURATION

4. WIND LESS THAN 2 KNOTS

5. TOTAL DIRECTIONAL CONTROL DISPLACEMENT IS 7.07 INCHES FROM FULL LEFT

6. SKID HEIGHT = 15 FEET

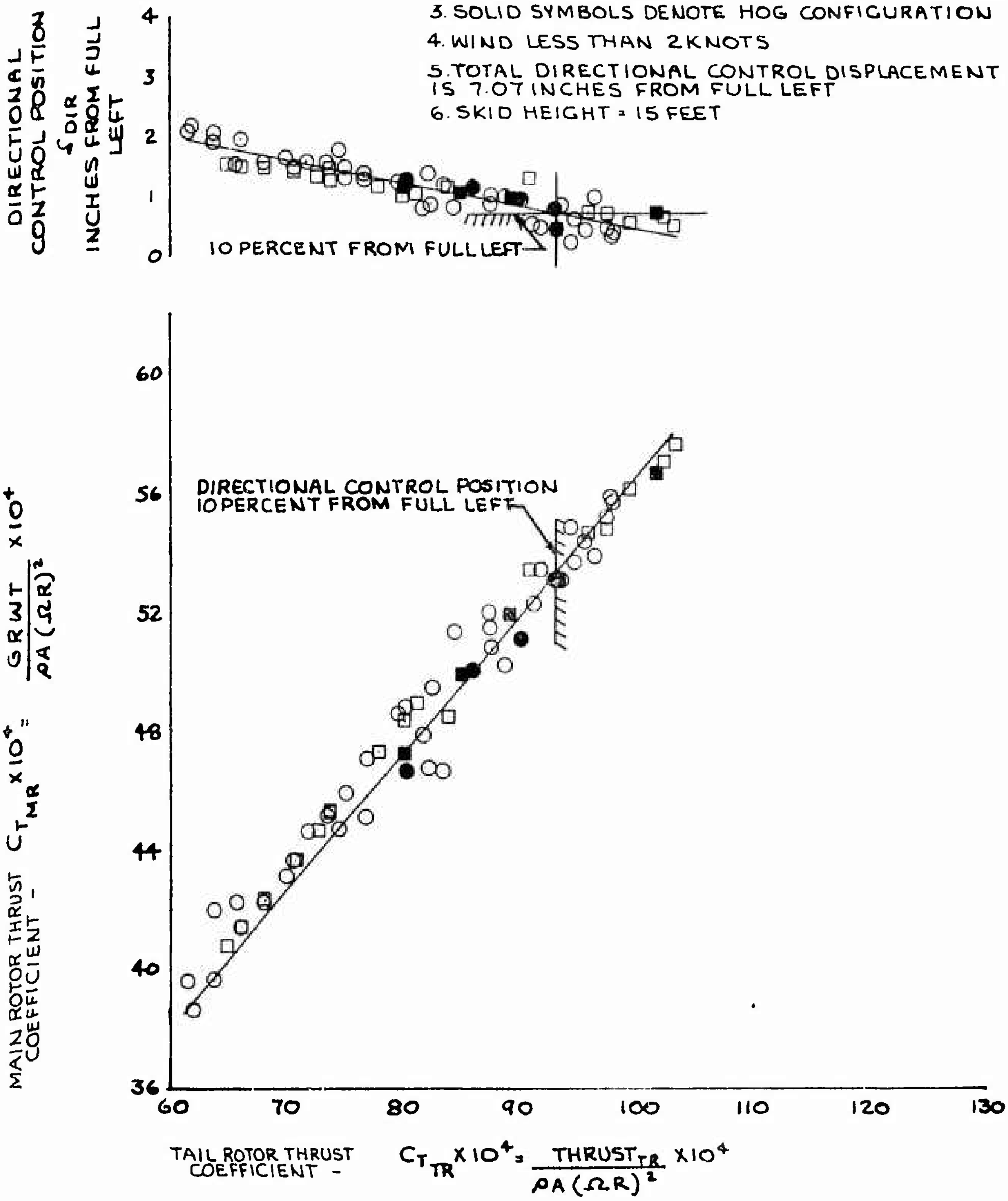


FIGURE NO. 6

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	ROTOR SPEED ~RPM
○	324
□	314

NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA

2. OPEN SYMBOLS DENOTE CLEAN CONFIGURATION
3. SOLID SYMBOLS DENOTE HOG CONFIGURATION
4. WIND LESS THAN 2 KNOTS
5. TOTAL DIRECTIONAL CONTROL DISPLACEMENT IS 7.07 INCHES FROM FULL LEFT
6. SKID HEIGHT = 30 FEET

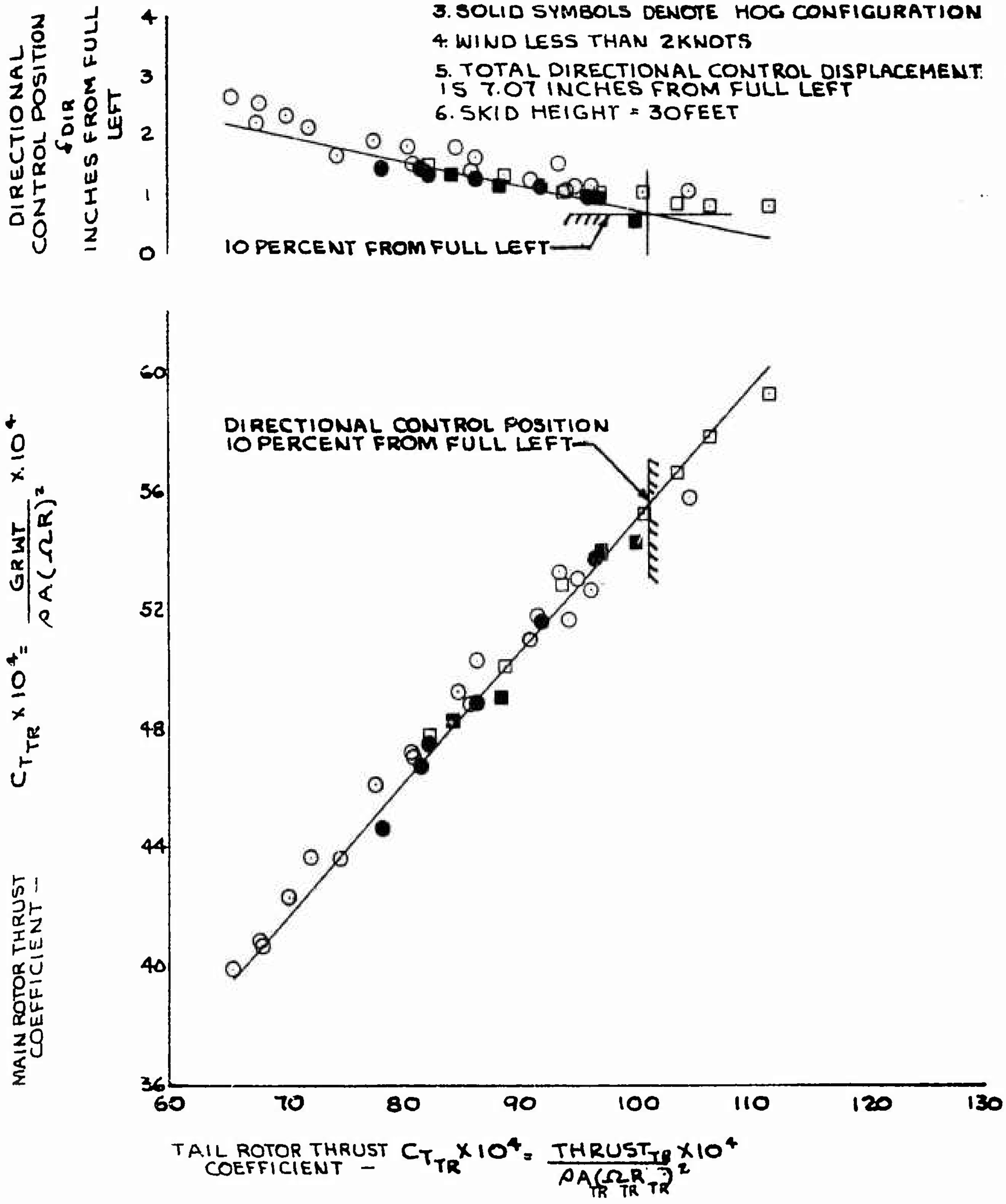


FIGURE NO. 7

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

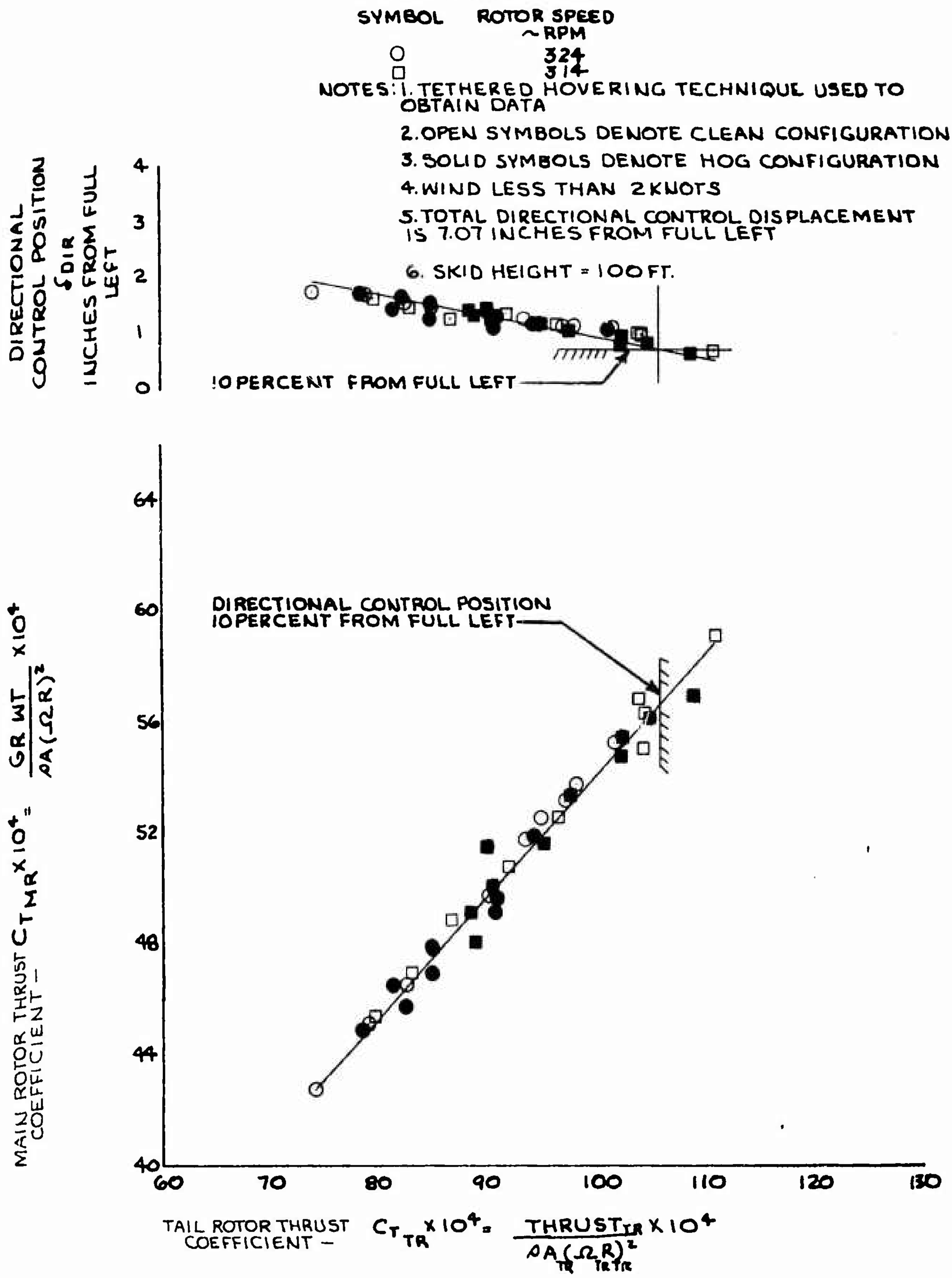


FIGURE NO 8

ANTITORQUE DRIVE SYSTEM HORSEPOWER IN A HOVER

AH-1G USA S/N 615247

DENSITY ALTITUDE = SEA LEVEL

NOTES: 1. TOTAL DIRECTIONAL CONTROL DISPLACEMENT IS 7.07 INCHES
2. FULL LEFT DIRECTIONAL CONTROL = 19° TAIL ROTOR PITCH
3. WIND LESS THAN 2 KNOTS
4. STANDARD DAY
5. MAIN ROTOR SPEED = 324 RPM

CURVE DERIVED FROM FIGURES 11 & 12 APP. VII

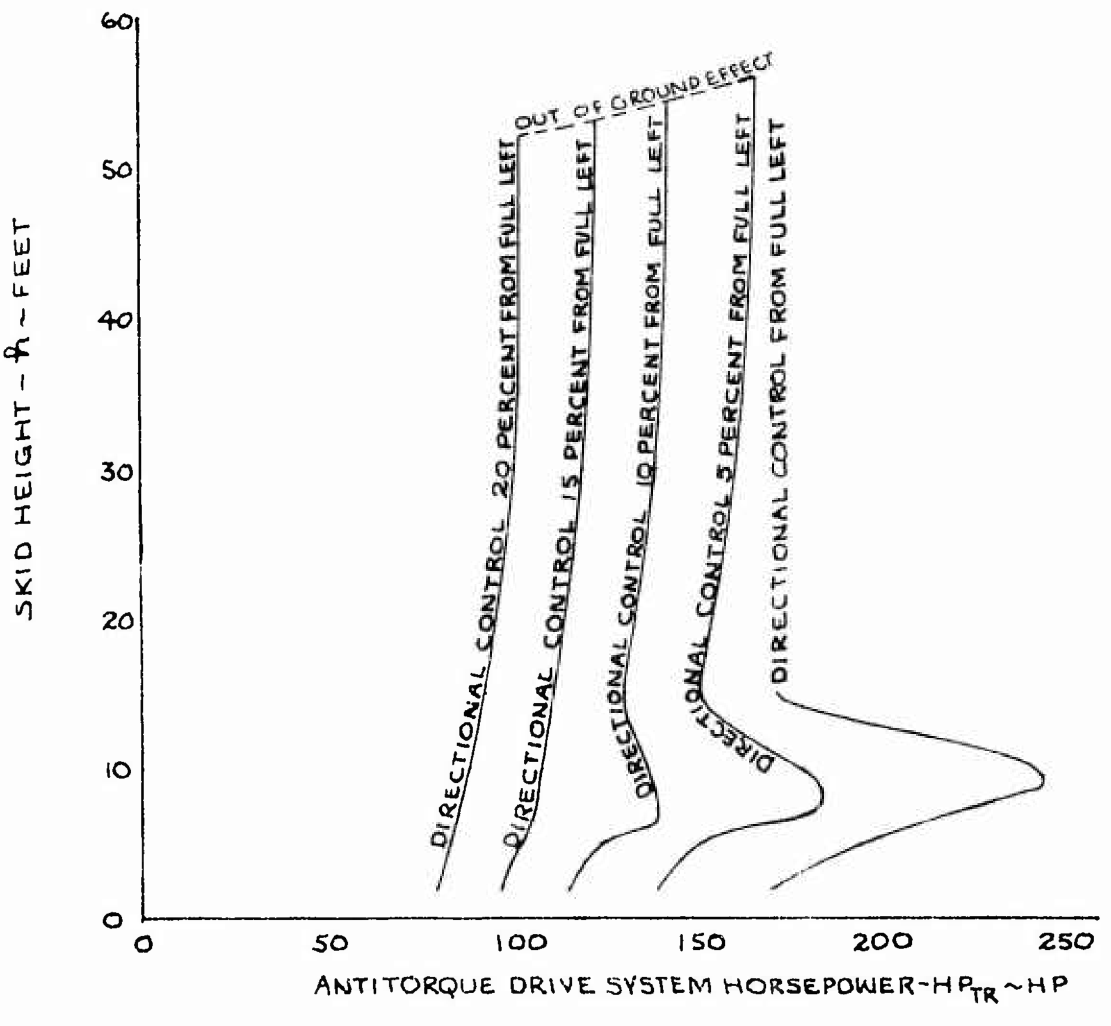


FIGURE NO. 9

ANTI-TORQUE ... SYSTEM HORSEPOWER IN A HOVER

... S247

... 5000 FT

NOTES: ... CONTROL DISPLACEMENT IS 7.07 INCHES ... CONTROL = 18° TAIL ROTOR PITCH

... RPM

CURVE ... 12, APP VII

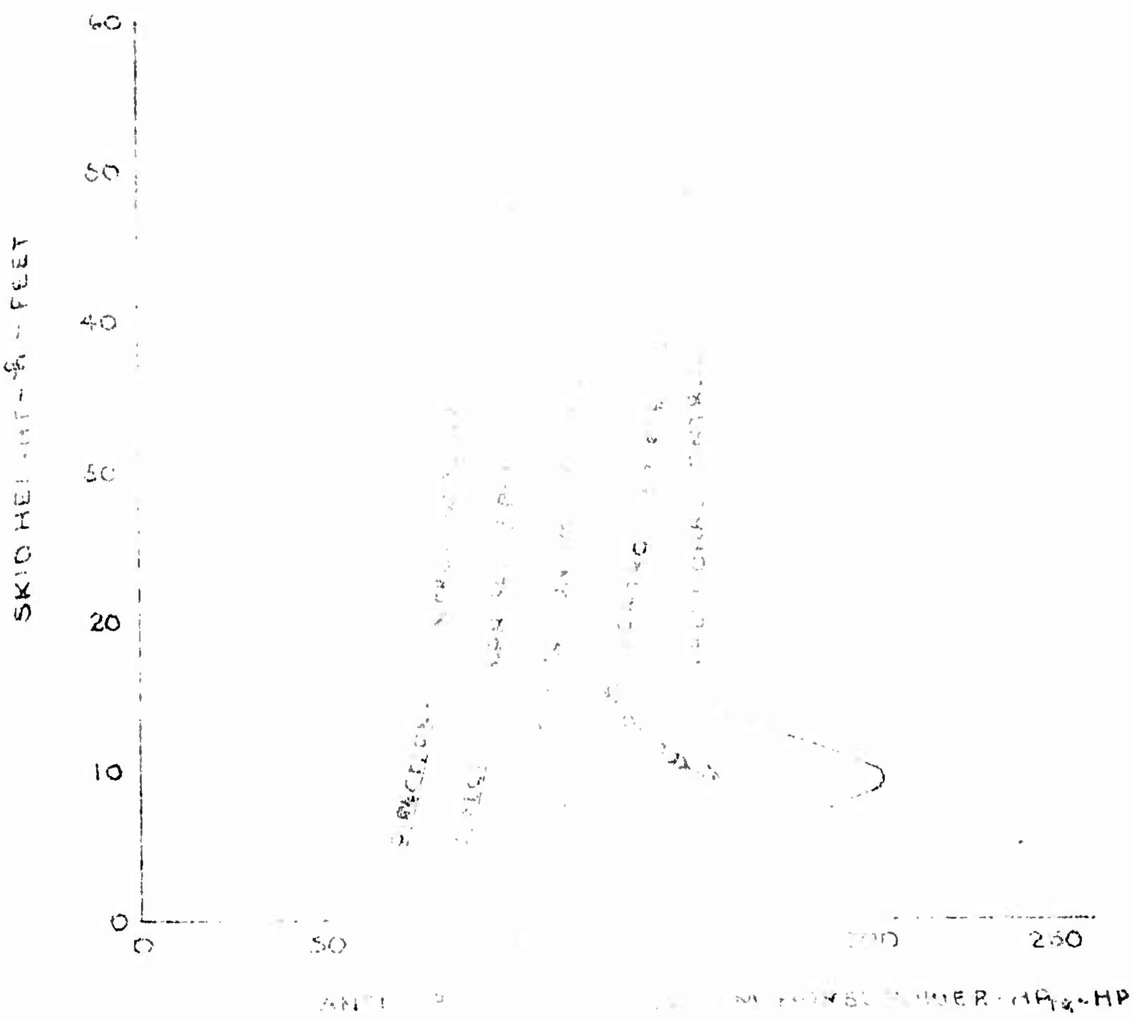


FIGURE No 10

ANTITORQUE DRIVE SYSTEM HORSEPOWER IN A HOVER

AH-1G USA S/N 615247

DENSITY ALTITUDE = 10000 FT.

NOTES: 1. TOTAL DIRECTIONAL CONTROL DISPLACEMENT IS 7.07 INCHES
2. FULL LEFT DIRECTIONAL CONTROL = 19° TAIL ROTOR PITCH
3. WIND LESS THAN 2 KNOTS
4. STANDARD DAY
5. MAIN ROTOR SPEED = 324 RPM

CURVE DERIVED FROM FIGURES 11 & 12 APP VII

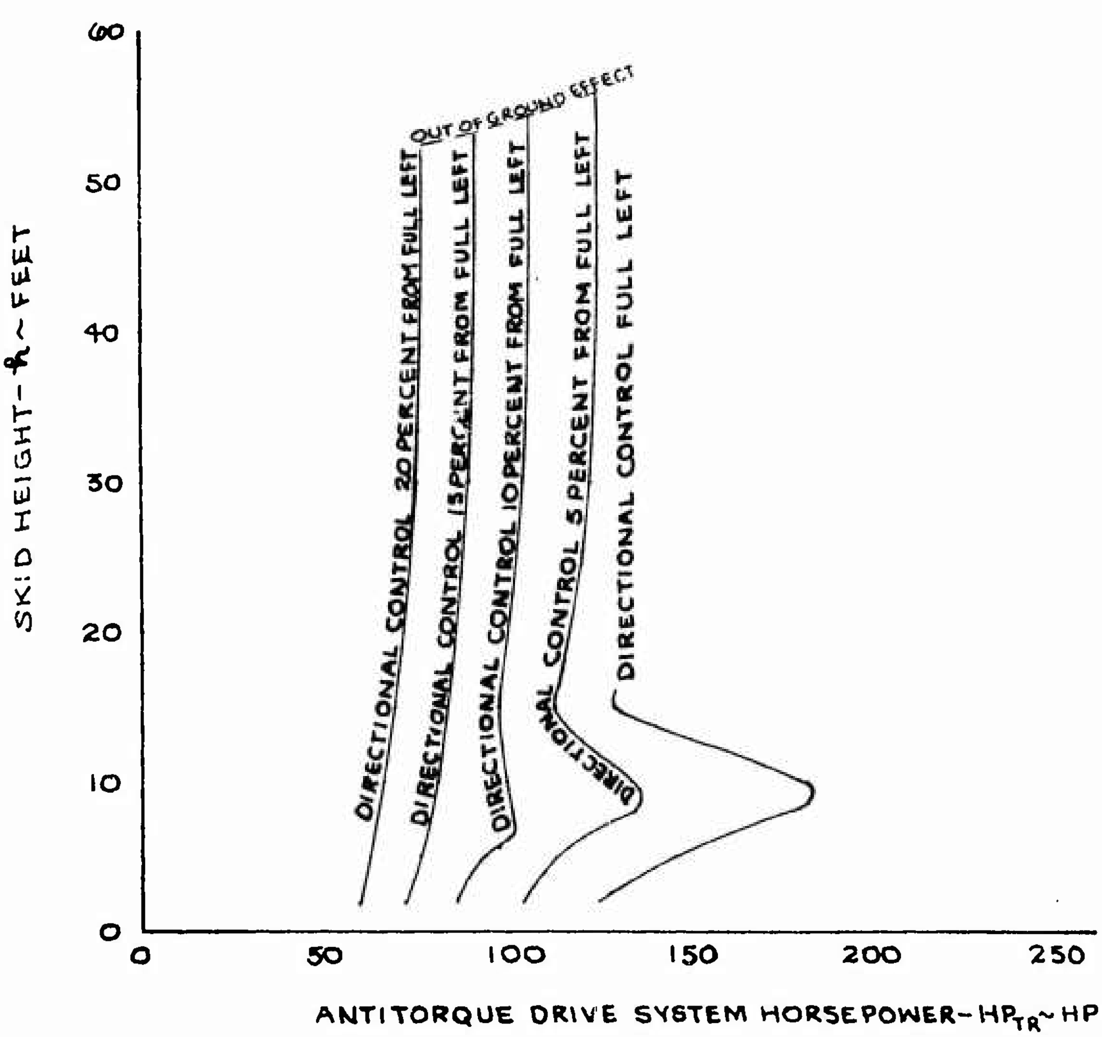


FIGURE NO. 11

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14-001

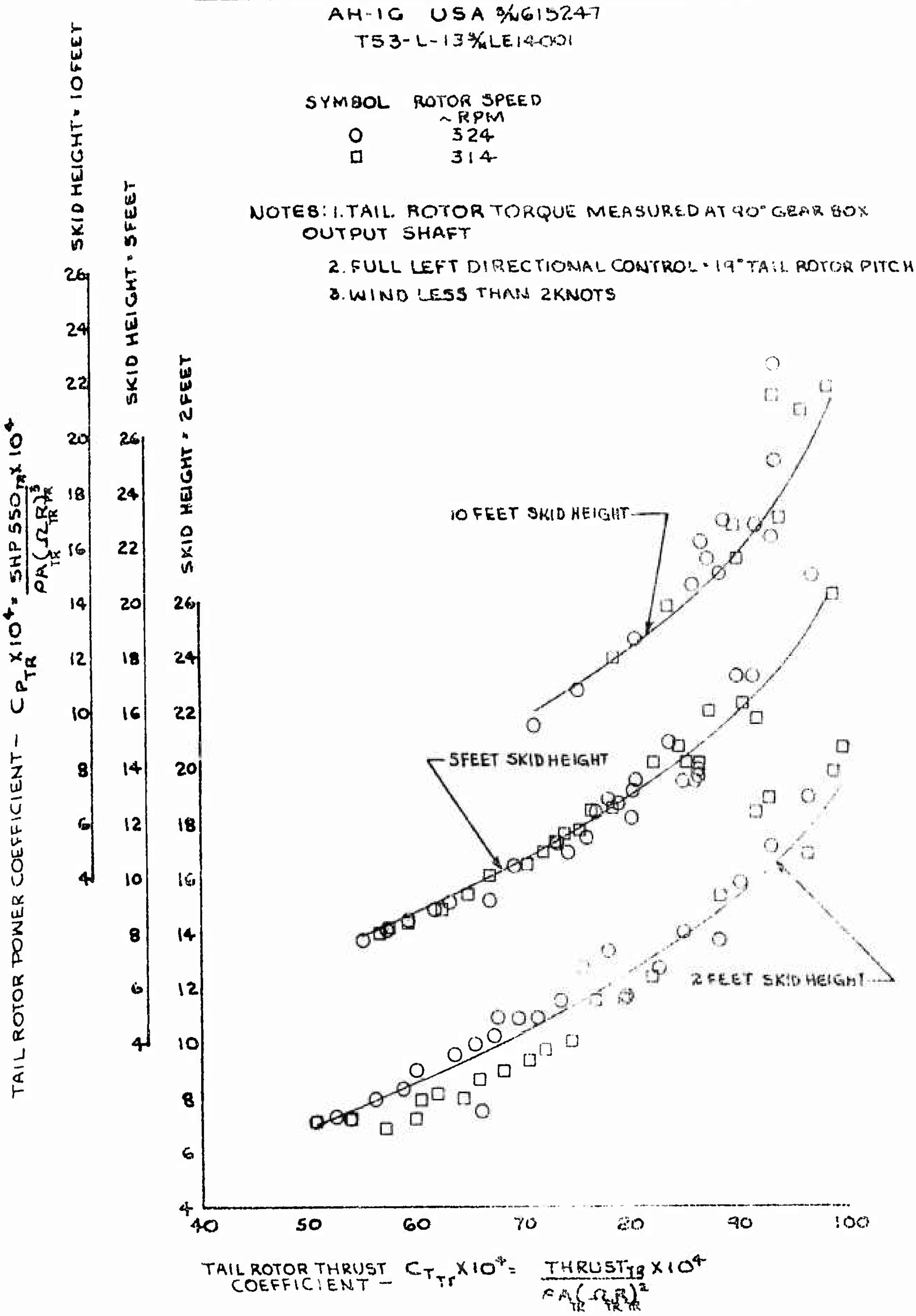


FIGURE NO. 12

NON DIMENSIONAL TAIL ROTOR PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	ROTOR SPEED ~RPM
○	324
□	314

NOTES: 1. TAIL ROTOR TORQUE MEASURED AT 90° GEAR BOX OUTPUT SHAFT

2. FULL LEFT DIRECTIONAL CONTROL = 19° TAIL ROTOR PITCH

3. WIND LESS THAN 2 KNOTS

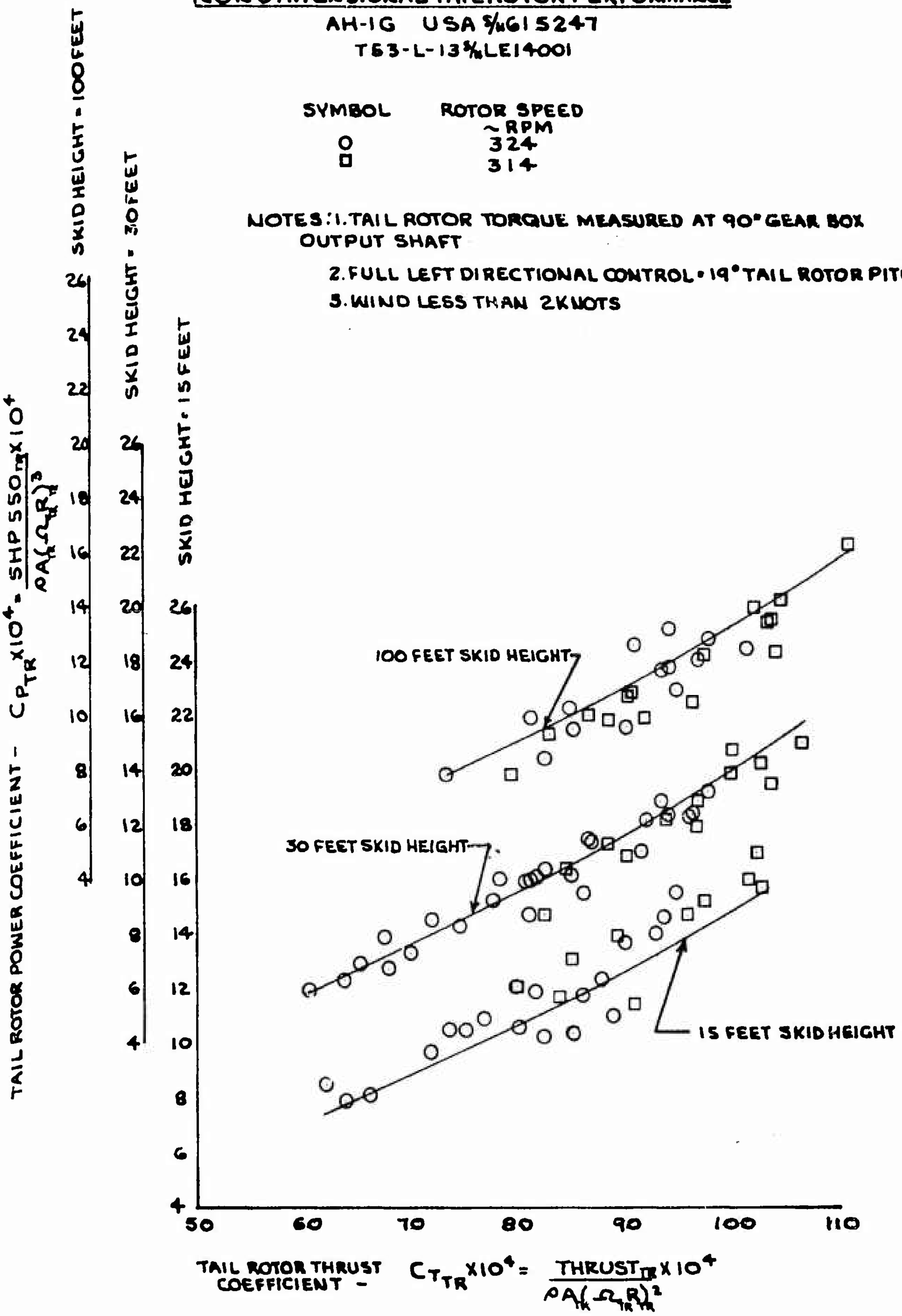


FIGURE NO. 13

OGE HOVERING PERFORMANCE

AH-1G USA S/N 615247

ENGINE PARTICLE SEPARATOR NOT INSTALLED

CONTRACT GUARANTEE COMPLIANCE CHECK

NOTES: 1. MAXIMUM POWER AVAILABLE BASED ON REFERENCE NO. 13 APP. I
2. AMBIENT TEMPERATURE = 35°C
3. WIND LESS THAN 2 KNOTS
4. ROTOR SPEED = 324 RPM

CURVE DERIVED FROM FIGURE NO. 19 APP. VII & REF. NO. 12 APP. I

FIGURE NO 14

OGE HOVERING PERFORMANCE

AH-1G USA S/N 615247

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVES BASED ON 10 PERCENT DIRECTIONAL CONTROL MARGIN OR MAXIMUM ENGINE POWER AVAILABLE, WHICHEVER IS LESS

2. WIND LESS THAN 2 KNOTS

3. ROTOR SPEED = 324 RPM

CURVES DERIVED FROM FIGURES 19 & 114 APP. VII

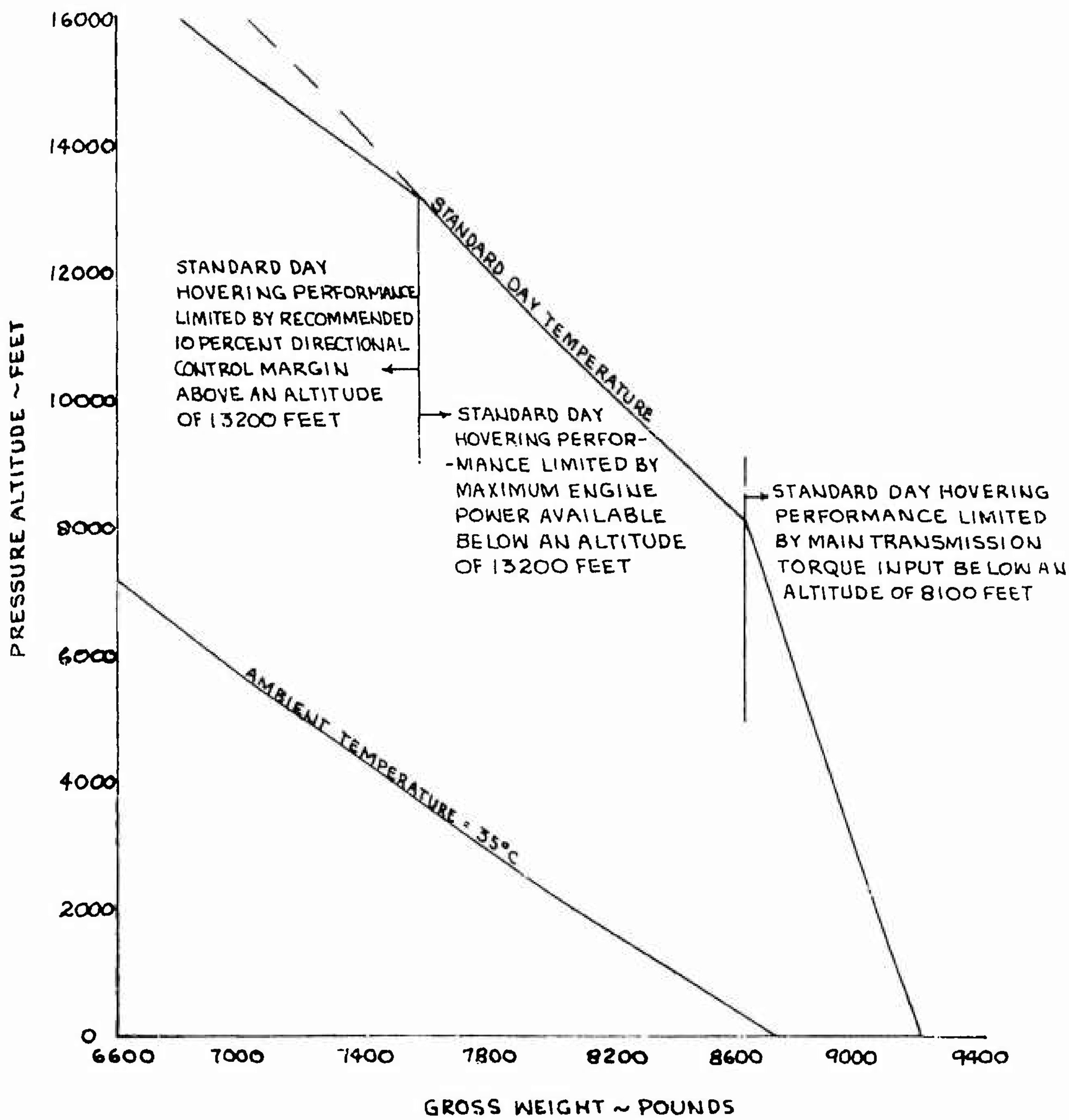


FIGURE NO. 15

IGE HOVERING PERFORMANCE

AH-1G USA S/N 615247

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. STANDARD DAY
2. WIND LESS THAN 2 KNOTS
3. SKID HEIGHT = 5 FEET
4. ROTOR SPEED = 324 RPM

CURVE DERIVED FROM FIGURES 1, 11, & 114 APP VII

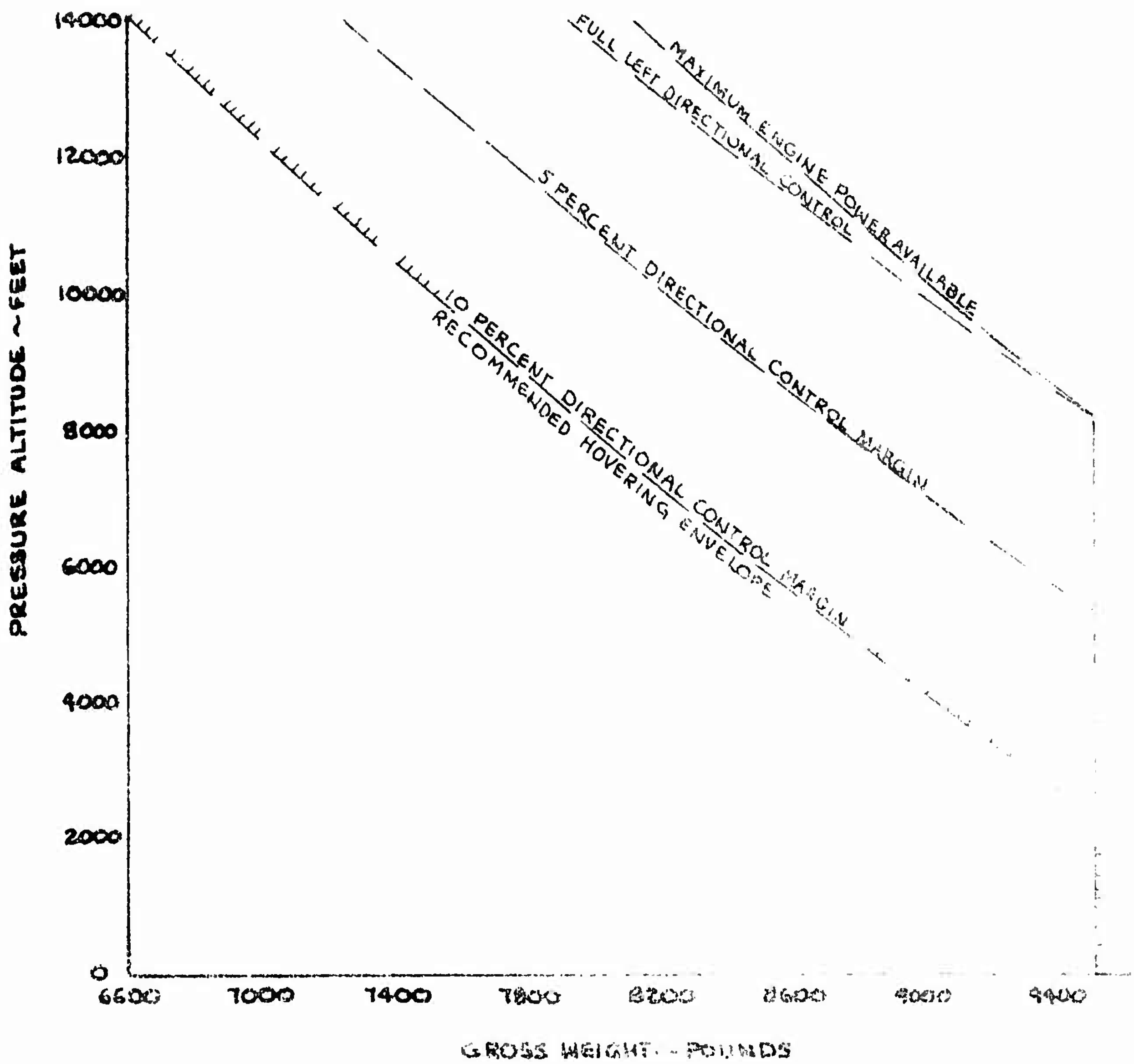
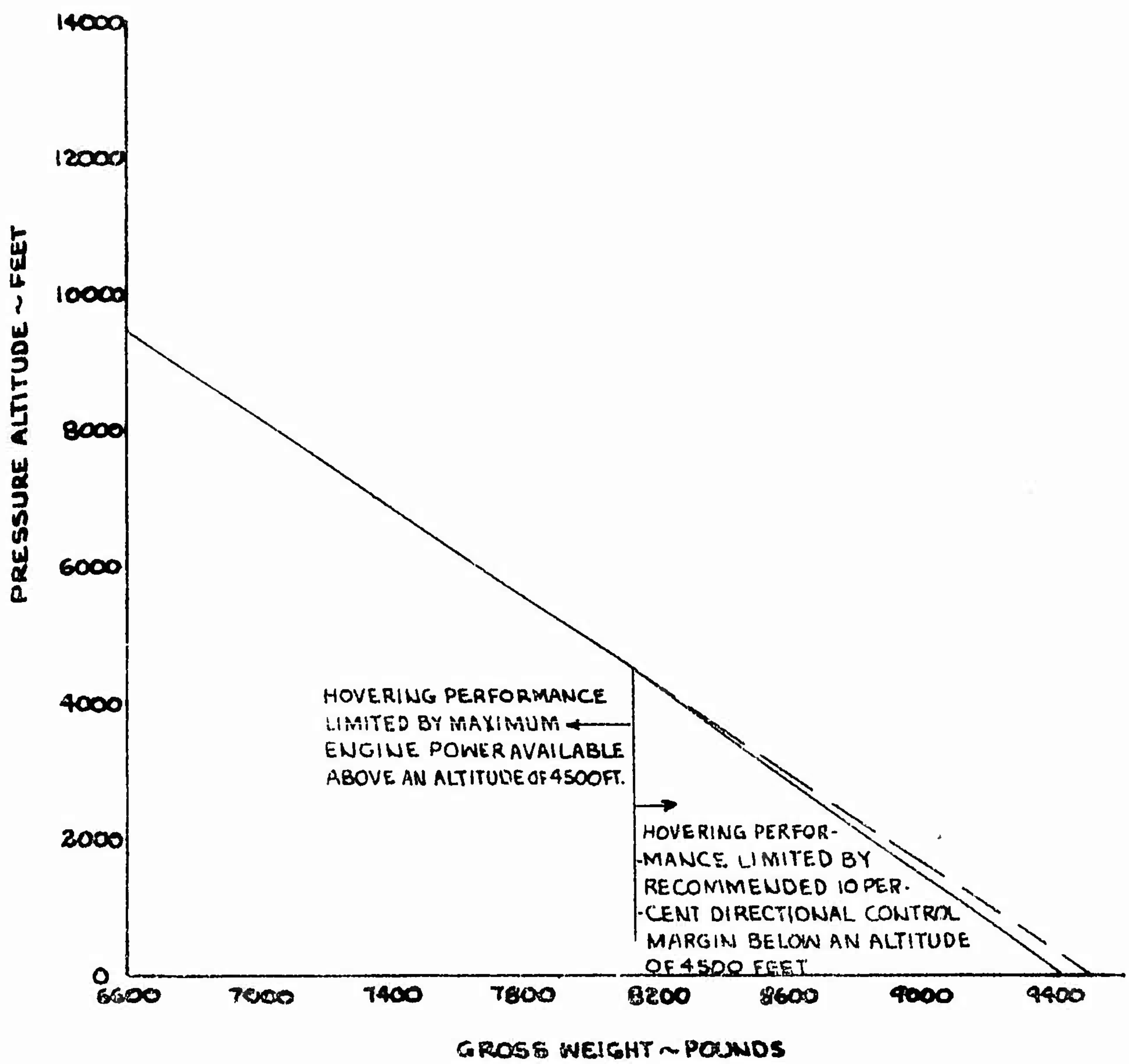


FIGURE No. 16
IGE HOVERING PERFORMANCE
AH-1G USA S/N 615247
ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. AMBIENT TEMPERATURE = 35°C
2. WIND LESS THAN 2 KNOTS
3. SKID HEIGHT = 5 FEET
4. ROTOR SPEED = 324 RPM
CURVES DERIVED FROM FIGURES: 1, 17, & 114 APP VII



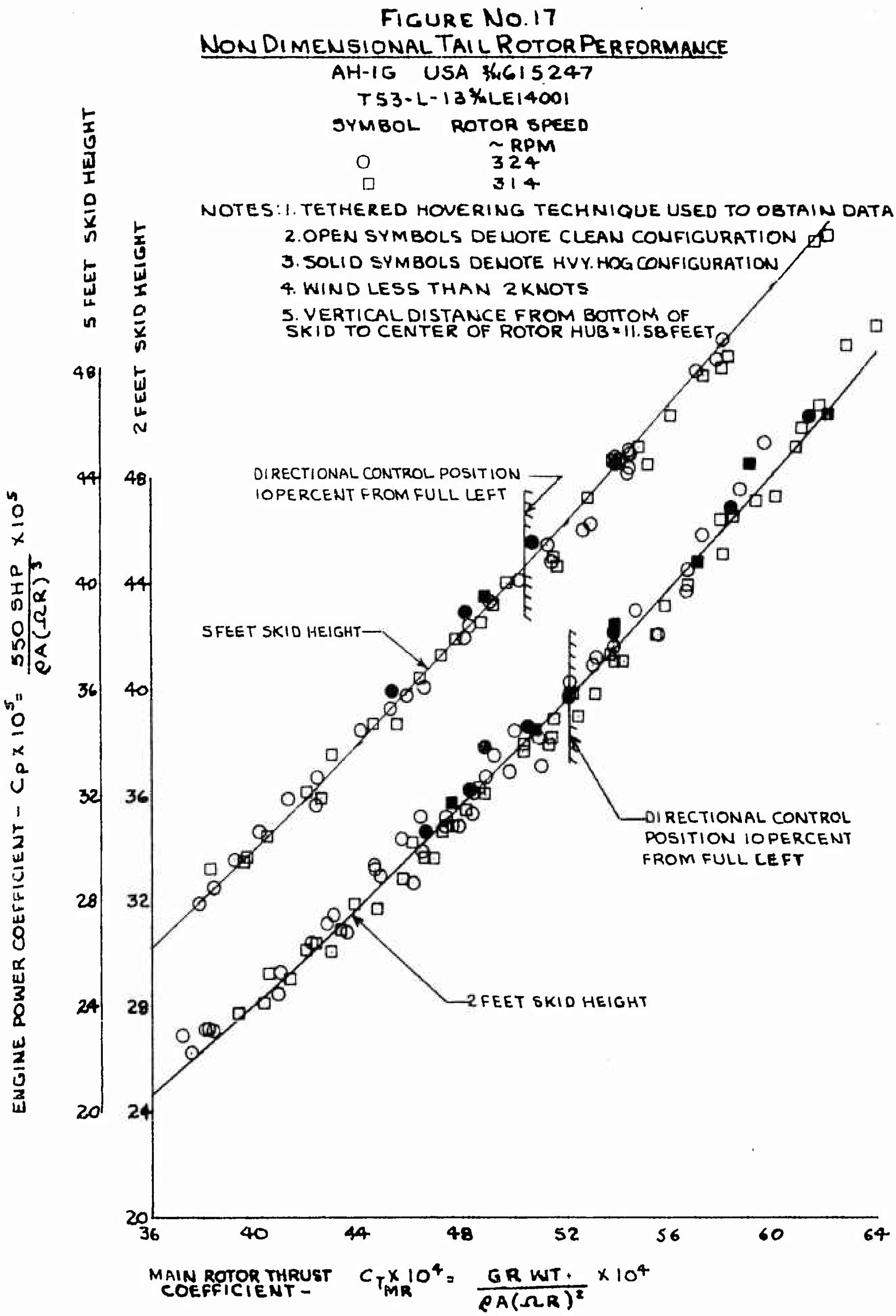
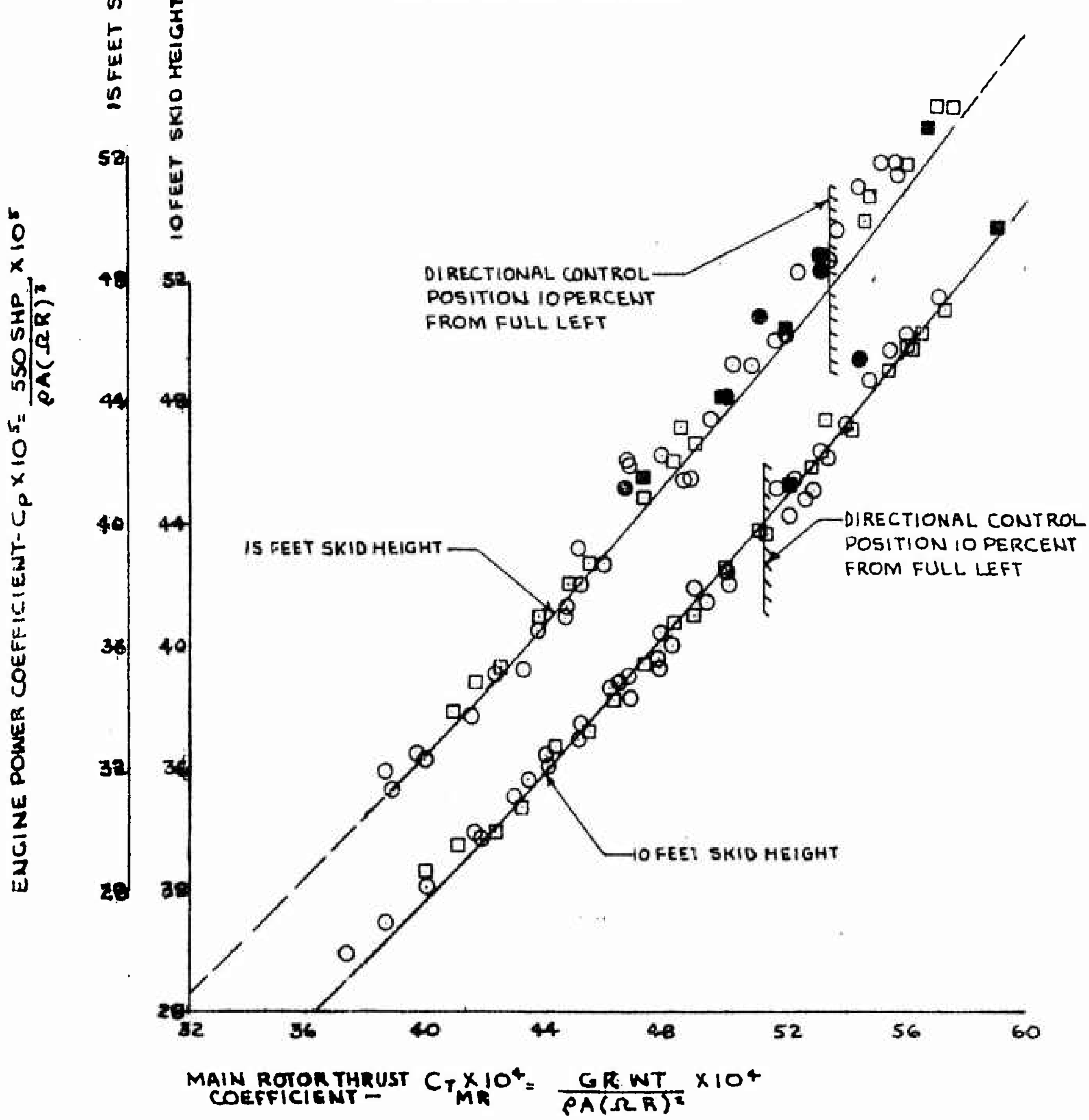


FIGURE No. 17
NON DIMENSIONAL TAIL ROTOR PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
SYMBOL ROTOR SPEED ~ RPM
○ 324
□ 314
NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. OPEN SYMBOLS DENOTE CLEAN CONFIGURATION
3. SOLID SYMBOLS DENOTE HVY. HOG CONFIGURATION
4. WIND LESS THAN 2 KNOTS
5. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 11.58 FEET
DIRECTIONAL CONTROL POSITION 10 PERCENT FROM FULL LEFT
5 FEET SKID HEIGHT
DIRECTIONAL CONTROL POSITION 10 PERCENT FROM FULL LEFT
2 FEET SKID HEIGHT
5 FEET SKID HEIGHT
2 FEET SKID HEIGHT
ENGINE POWER COEFFICIENT - $C_P \times 10^5 = \frac{550\ SHP}{\rho A(\Omega R)^3} \times 10^5$
MAIN ROTOR THRUST COEFFICIENT - $C_{T_{MR}} \times 10^4 = \frac{GR\ WT.}{\rho A(\Omega R)^2} \times 10^4$

FIGURE NO. 18
NON DIMENSIONAL HOVERING PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYMBOL	ROTOR SPEED ~ RPM
○	324
□	314

NOTES: 1. TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. OPEN SYMBOLS DENOTE CLEAN CONFIGURATION
3. SOLID SYMBOLS DENOTE HVY. HOG CONFIGURATION
4. WIND LESS THAN 2 KNOTS
5. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 11.58 FEET



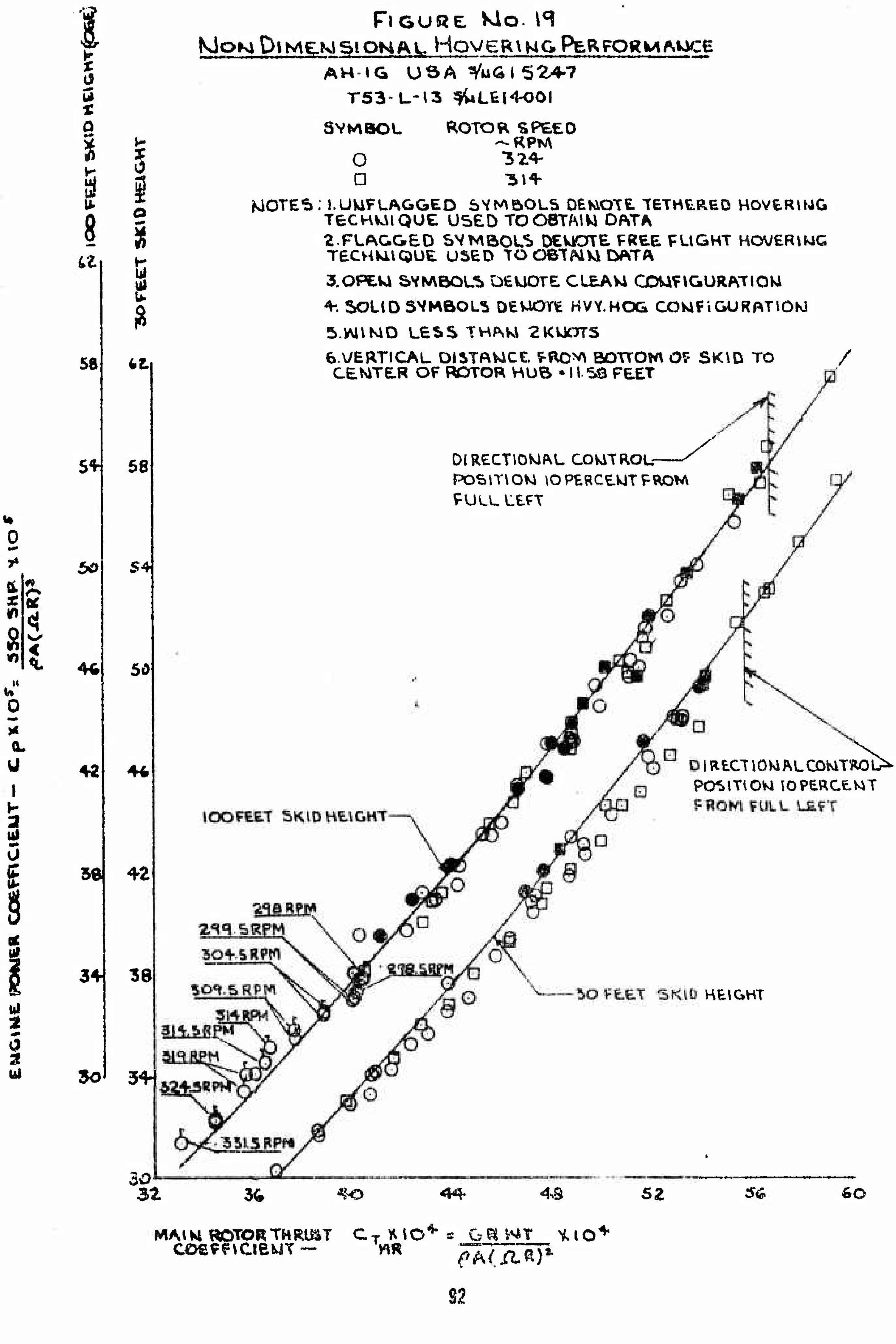


FIGURE No. 19
NON DIMENSIONAL HOVERING PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
SYMBOL ROTOR SPEED ~RPM
324
314
NOTES: 1. UNFLAGGED SYMBOLS DENOTE TETHERED HOVERING TECHNIQUE USED TO OBTAIN DATA
2. FLAGGED SYMBOLS DENOTE FREE FLIGHT HOVERING TECHNIQUE USED TO OBTAIN DATA
3. OPEN SYMBOLS DENOTE CLEAN CONFIGURATION
4. SOLID SYMBOLS DENOTE HVY. HOG CONFIGURATION
5. WIND LESS THAN 2 KNOTS
6. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 11.58 FEET
DIRECTIONAL CONTROL POSITION 10 PERCENT FROM FULL LEFT
DIRECTIONAL CONTROL POSITION 10 PERCENT FROM FULL LEFT
100 FEET SKID HEIGHT
30 FEET SKID HEIGHT
298 RPM
299.5 RPM
304.5 RPM
309.5 RPM
314 RPM
314.5 RPM
319 RPM
324.5 RPM
331.5 RPM
298.5 RPM
100 FEET SKID HEIGHT (OGE)
30 FEET SKID HEIGHT
ENGINE POWER COEFFICIENT – $C_P \times 10^5 = \frac{550\ SHP \times 10^5}{\rho A (\Omega R)^3}$
MAIN ROTOR THRUST COEFFICIENT – $C_{T_{MR}} \times 10^4 = \frac{GR\ WT}{\rho A (\Omega R)^2} \times 10^4$
30
34
38
42
46
50
54
58
62
32
36
40
44
48
52
56
60

FIGURE NO 20

HOVERING IN WIND ENVELOPE FOR A
TEN PERCENT DIRECTIONAL CONTROL MARGIN

AH-1G T53-L-13

SKID HEIGHT = 7 FEET

NOTES: 1. CURVES DERIVED FROM FIGURE NO. 21 APP. VII
2. WIND VELOCITY PRESENTED FOR CRITICAL WIND AZIMUTH
3. SEVEN FOOT SKID HEIGHT REPRESENTS MOST CRITICAL CONDITION
4. FULL LEFT DIRECTIONAL CONTROL = 19° DEGREES TAIL ROTOR BLADE ANGLE
5. 10 PERCENT DIRECTIONAL CONTROL REMAINING FROM MEAN CONTROL POSITION REQUIRED DURING STABILIZED HOVER
6. YAW SCAS OFF
7. STANDARD DAY

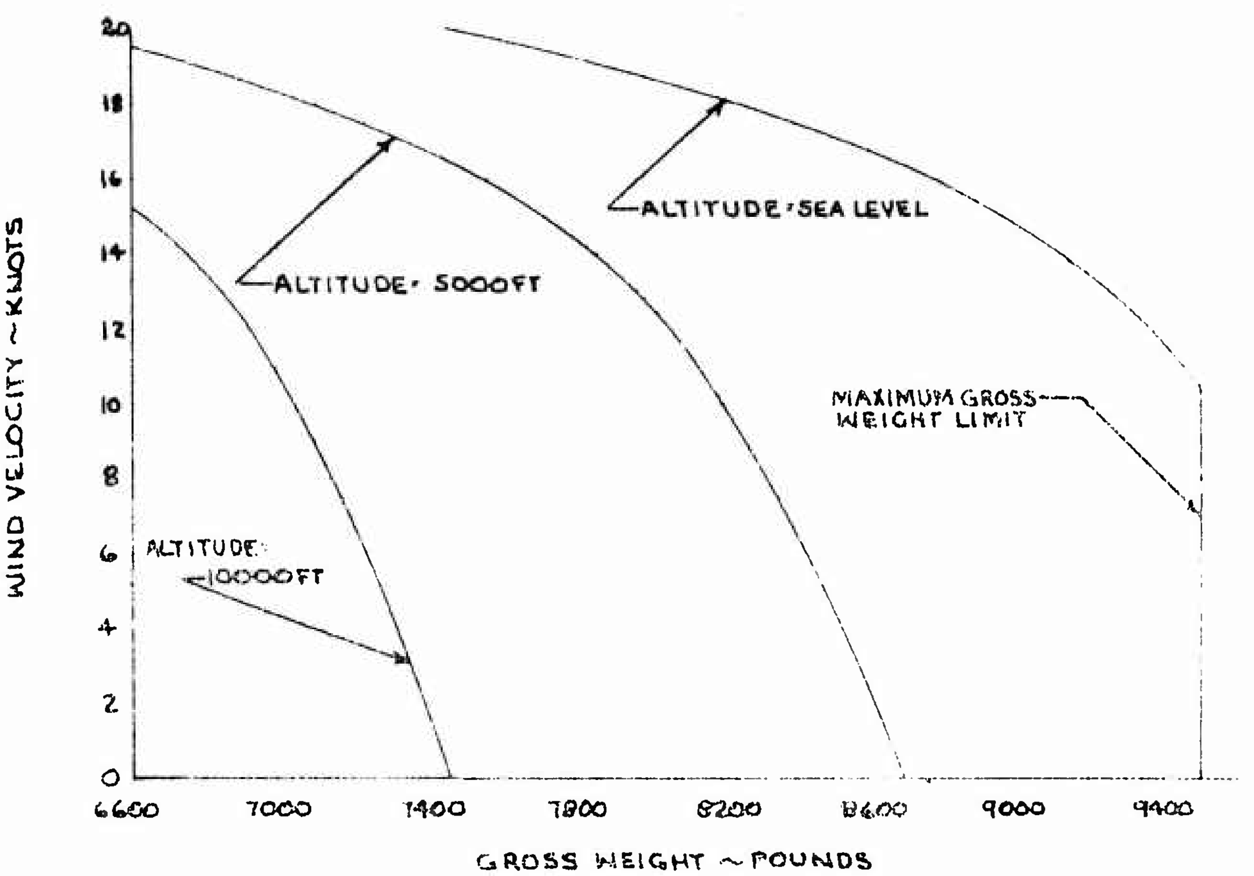


FIGURE NO 21
IGE HOVERING IN WIND
FOR A TEN PERCENT DIRECTIONAL CONTROL MARGIN

AH-1G USA S/N 615247
T53-L-13 S/LE14001
SKID HEIGHT = 7 FEET

NOTES: 1. POINTS DERIVED FROM NO. 22 THROUGH 25 APP. VII
2. WIND VELOCITY PRESENTED FOR CRITICAL WIND AZIMUTH
3. SEVEN FOOT SKID HEIGHT REPRESENTS MOST CRITICAL CONDITION
4. FULL LEFT DIRECTIONAL CONTROL = 19° DEGREES TAIL ROTOR BLADE ANGLE
5. 10 PERCENT DIRECTIONAL CONTROL REMAINING FROM MEAN CONTROL POSITION REQUIRED DURING STABILIZED HOVER
6. YAW SCAS OFF
7. STANDARD DAY

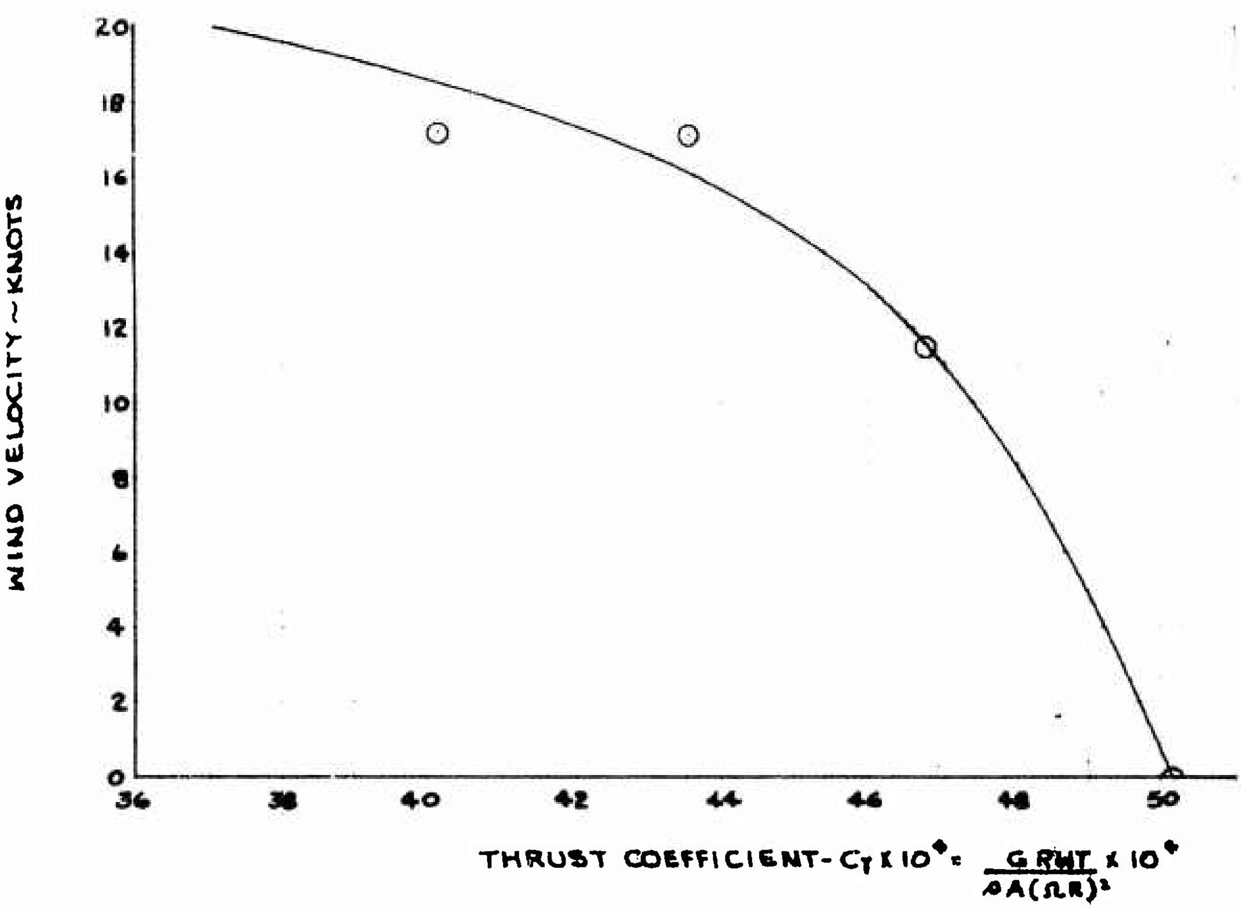


FIGURE NO. 22

DIRECTIONAL CONTROL SUMMARY

AH-1G USA S/N 615247

HVY SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

ALTITUDE H_D ~ FT	GROSS WEIGHT ~ LB	LONG. C.G ~ IN.	ROTOR SPEED ~ RPM	THRUST COEFF
140	8060	200.4 (AFT)	324	0.004018

NOTES:

1. 10% DIRECTIONAL CONTROL REMAINING FROM MEAN CONTROL POSITION REQUIRED DURING STABILIZED FLIGHT CONDITION
2. YAW SCAS OFF
3. TOTAL DIRECTIONAL CONTROL DISPLACEMENT = 7.07 IN FROM FULL LEFT
4. SHADED AREAS REPRESENT LESS THAN 10% DIRECTIONAL CONTROL MARGIN
5. POINTS DERIVED FROM REFERENCE 15 APP I

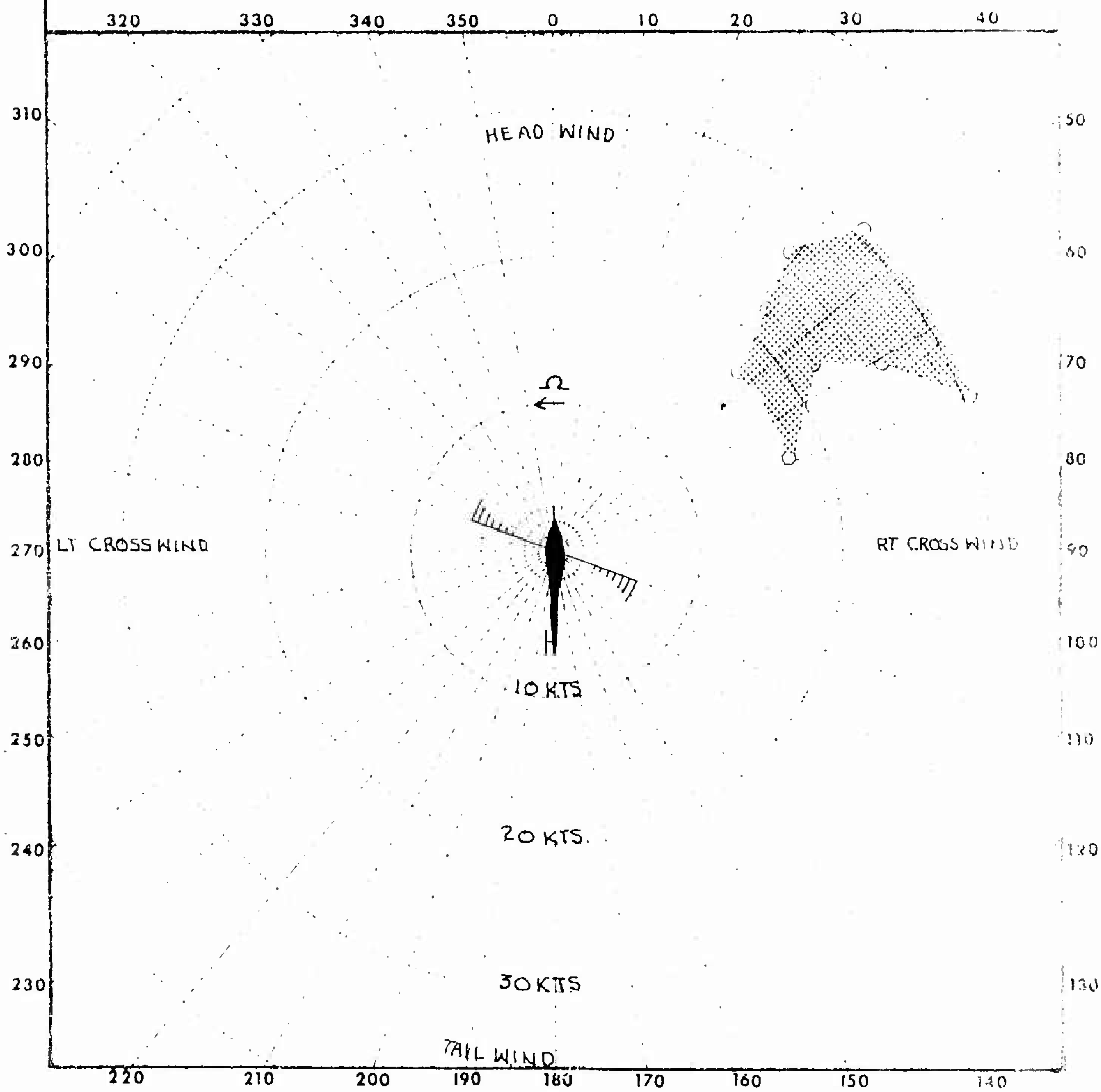


FIGURE NO. 23

DIRECTIONAL CONTROL SUMMARY

AH-1G USA S/N 615247

HVY SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

ALTITUDE H_D ~ FT.	GROSS WEIGHT ~ LB.	LONG. C.G. ~ IN.	ROTOR SPEED ~ RPM	THRUST COEFF ~ C_T
-40	8245	199.6 (AFT)	314	0.004355

NOTES:

1. 10% DIRECTIONAL CONTROL REMAINING FROM MEAN CONTROL POSITION REQUIRED DURING STABILIZED FLIGHT CONDITION
2. YAW SCAS OFF
3. TOTAL DIRECTIONAL CONTROL DISPLACEMENT 7.07 IN. FROM FULL LEFT
4. SHADED AREA REPRESENTS LESS THAN 10% DIRECTIONAL CONTROL MARGIN
5. POINTS DERIVED FROM REFERENCE 15 APP I

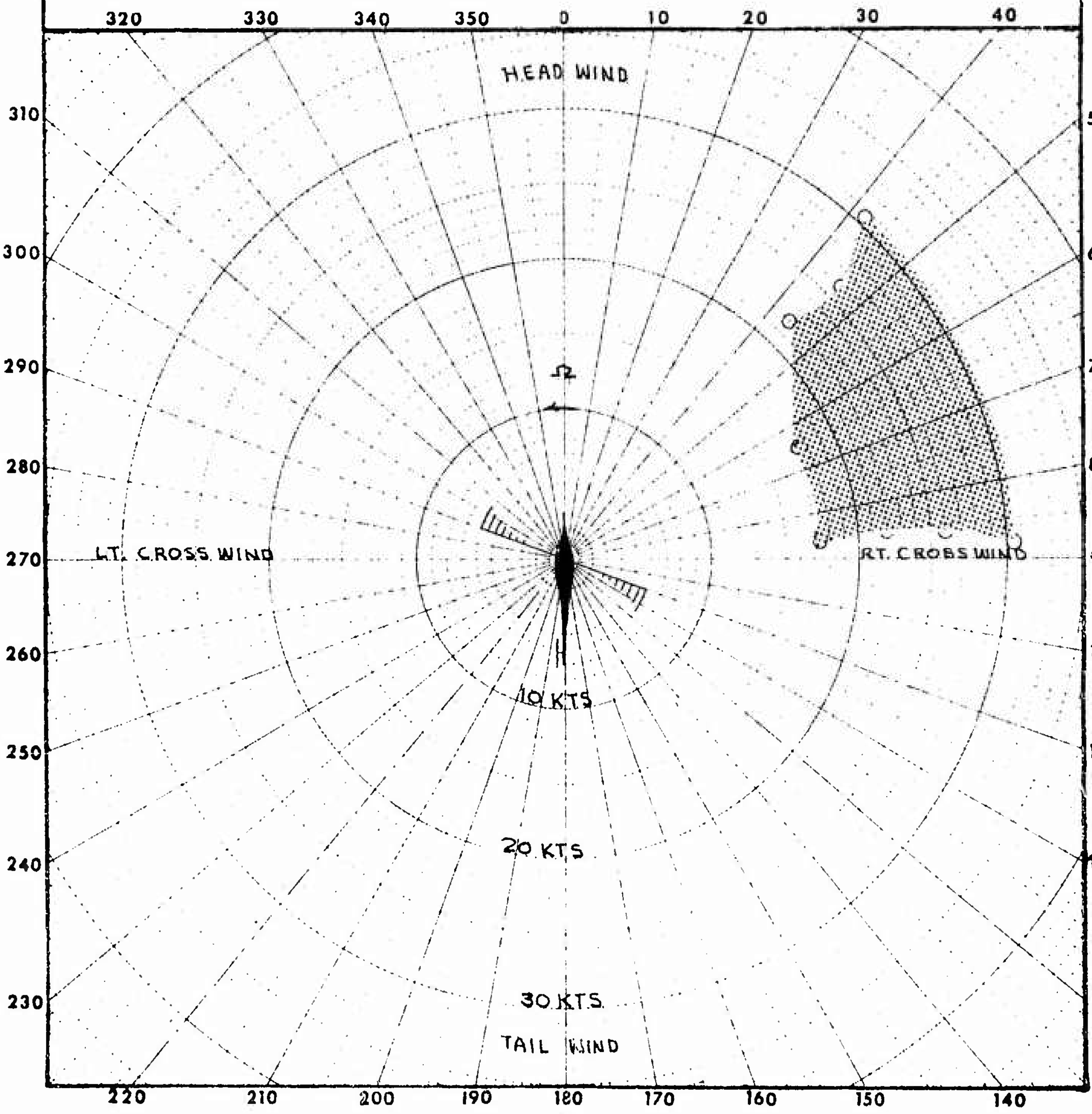


FIGURE No 24

DIRECTIONAL CONTROL SUMMARY

AH-1G USA S/N 615247

HVY SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

ALTITUDE H_D ~ FT.	GROSS WEIGHT ~ LB.	LONG C G ~ IN.	ROTOR SPEED ~ RPM	THRUST COEFF ~ C_T
5270	8050	200.7(AFT)	324	0.004678

NOTES.

1. 10% DIRECTIONAL CONTROL REMAINING FROM MEAN CONTROL POSITION REQUIRED DURING STABILIZED FLIGHT CONDITION
2. YAW SCAS OFF
3. TOTAL DIRECTIONAL CONTROL DISPLACEMENT = 7.07 IN FROM FULL LEFT
4. SHADED AREA REPRESENTS LESS THAN 10% DIRECTIONAL CONTROL MARGIN
5. POINTS DERIVED FROM REFERENCE 15 APP I

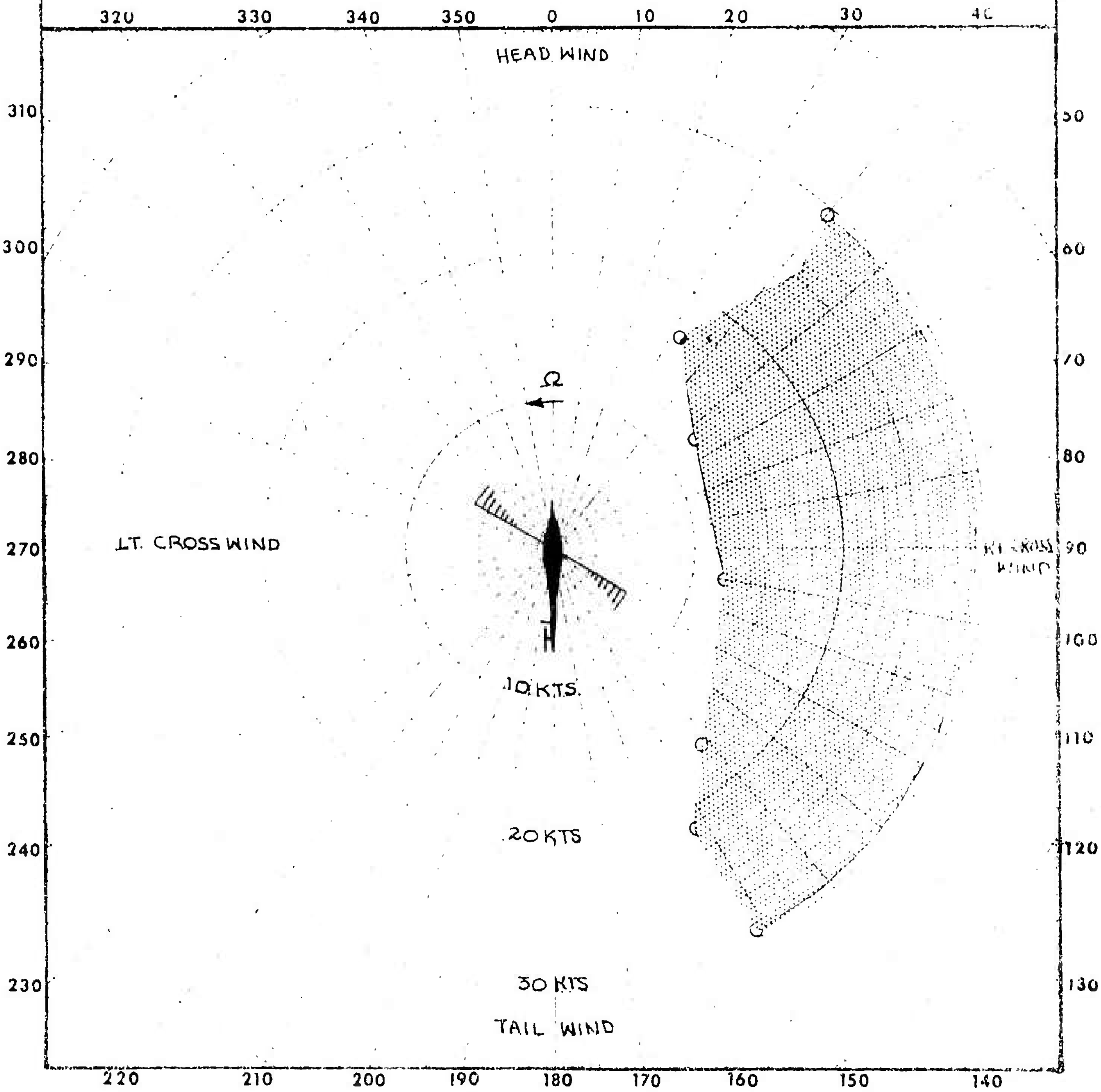


FIGURE NO 25

DIRECTIONAL CONTROL SUMMARY

AH-1G USA S/N 615247

HVY SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

ALTITUDE H_D ~ FT	GROSS WEIGHT ~ LB	LONG CG ~ IN.	ROTOR SPEED ~ RPM	THRUST COEFF ~ C_T
11120	7210	195.4 (MID)	324	0.005023

NOTES:

1. 10% DIRECTIONAL CONTROL REMAINING FROM MEAN CONTROL POSITION REQUIRED DURING STABILIZED FLIGHT CONDITION
2. YAW SCAS OFF
3. TOTAL DIRECTIONAL CONTROL DISPLACEMENT 7.07 IN. FROM FULL LEFT
4. SHADED AREA REPRESENTS LESS THAN 10% DIRECTIONAL CONTROL MARGIN
5. POINTS DERIVED FROM REFERENCE 15 APP I

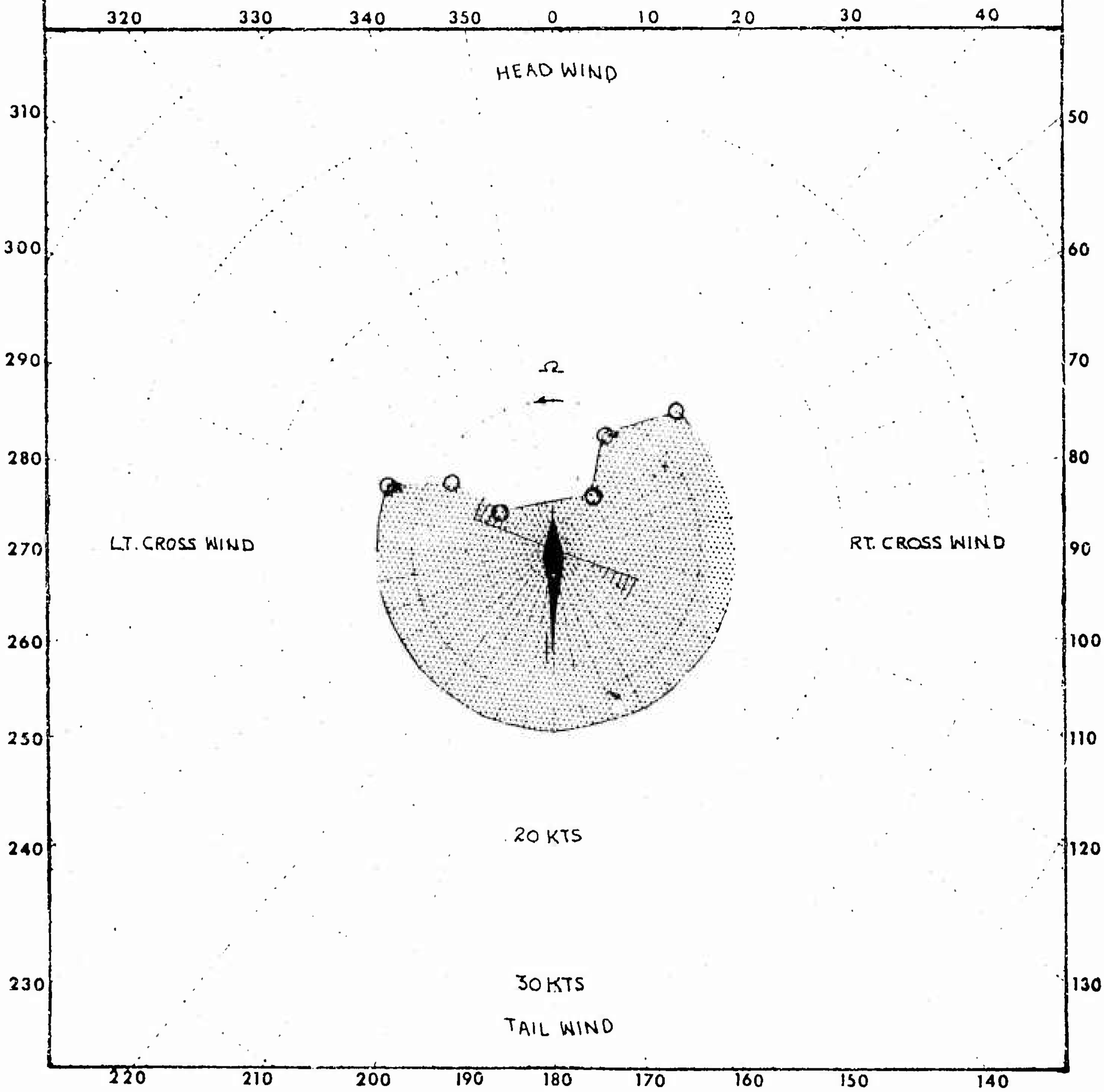


FIGURE NO. 26

TAKE OFF PERFORMANCE SUMMARY

AH-1G T53-L-13

HVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY = 195 IN. (MID)

TECHNIQUE = LEVEL FLIGHT ACCELERATION FROM A 3 FOOT HOVER

NOTES: 1. DIRECTIONAL CONTROL MARGIN OF 10 PERCENT BASED ON FIGURE NO. 1 APP VII

2. CURVES DERIVED FROM FIGURE NOS. 17, 18, 27 & 114 & 131 APP VII

3. STANDARD DAY CONDITIONS

4. ROTOR SPEED = 324 RPM

5. WIND VELOCITY ≤ 3 KNOTS

6. V_{IND} = INDICATED AIRSPEED DURING CLIMB-OUT

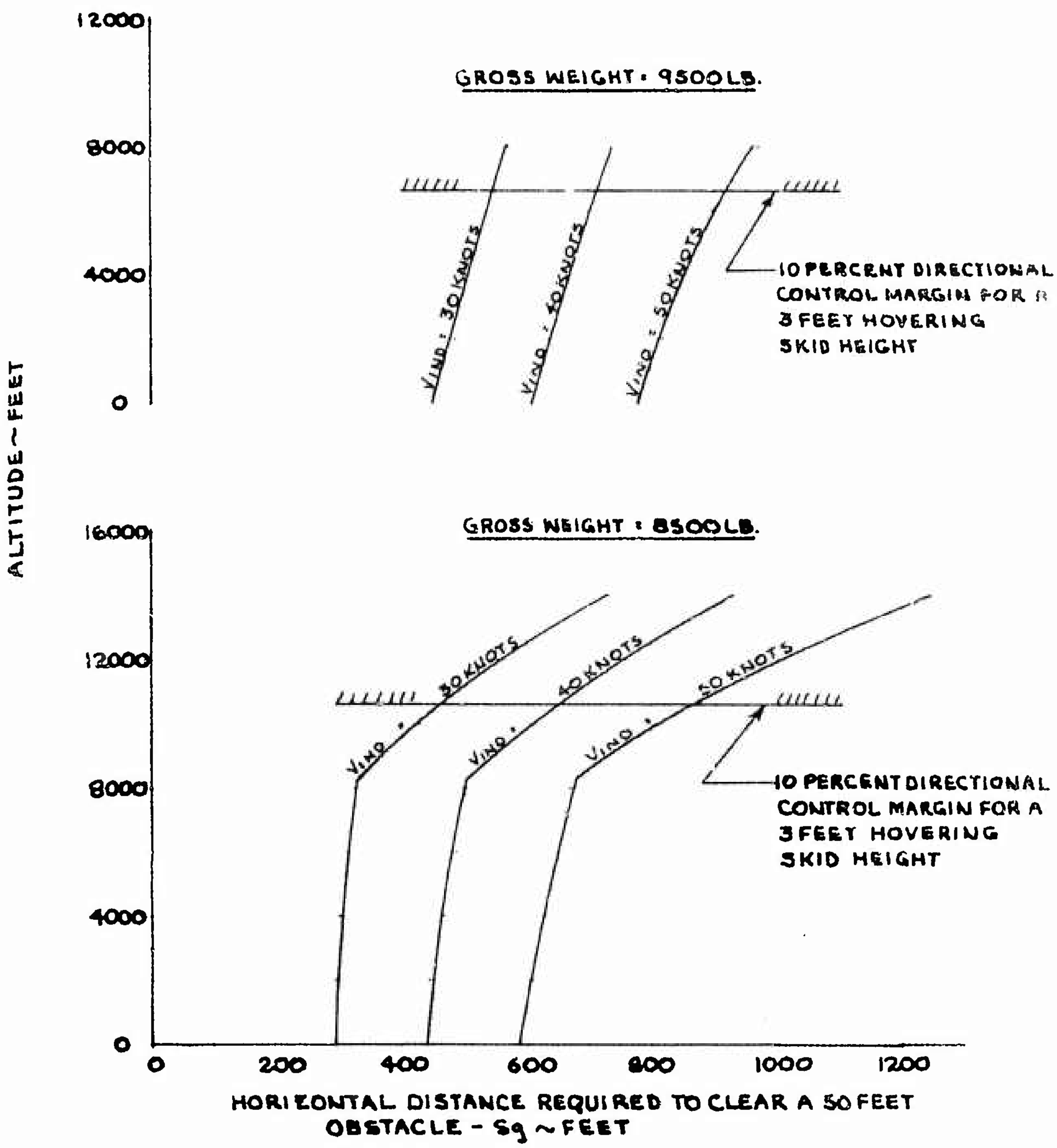


FIGURE NO. 27
VARIATION OF TAKE OFF DISTANCE
WITH AIRSPEED AT 50 FEET AND ΔC_P
AH-1G USA S/N 615247
HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

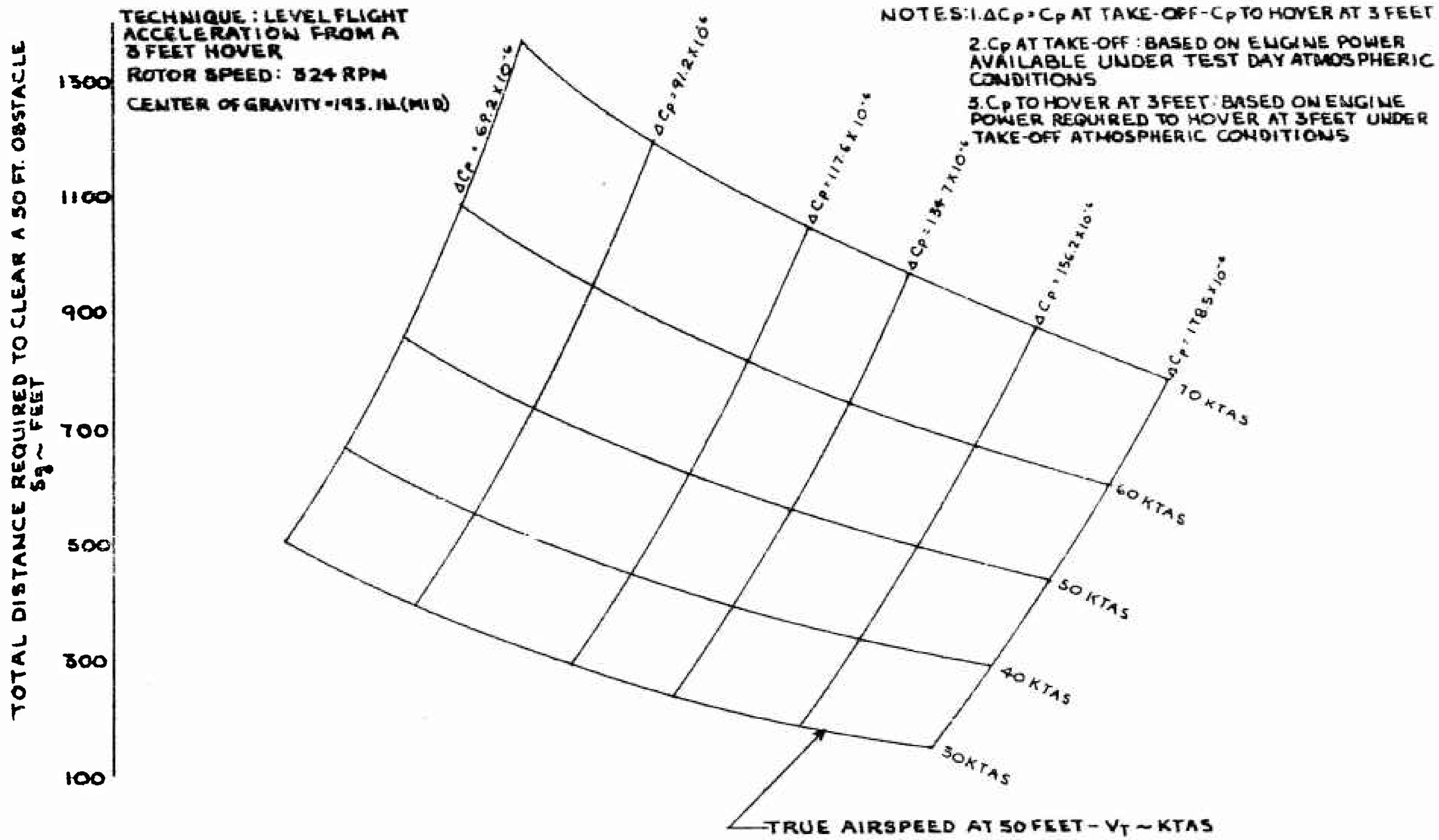


FIGURE NO 28

TAKE-OFF DISTANCE REQUIRED TO CLEAR A 50 FEET OBSTACLE

AH-1G USA S/N 615247

HVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED
CENTER OF GRAVITY [illegible] INCHES (MID)

TECHNIQUE: LEVEL FLIGHT ACCELERATION FROM A THREE FEET HOVER

NOTE: 1. ΔCp = Cp AT 50 FEET − Cp TO HOVER AT 3 FEET
2. Cp AT 50 FEET BASED ON ENGINE POWER AVAILABLE UNDER TEST DAY ATMOSPHERIC CONDITIONS
3. Cp TO HOVER AT 3 FEET BASED ON ENGINE POWER REQUIRED TO HOVER AT 3 FEET UNDER TAKE-OFF ATMOSPHERIC CONDITIONS

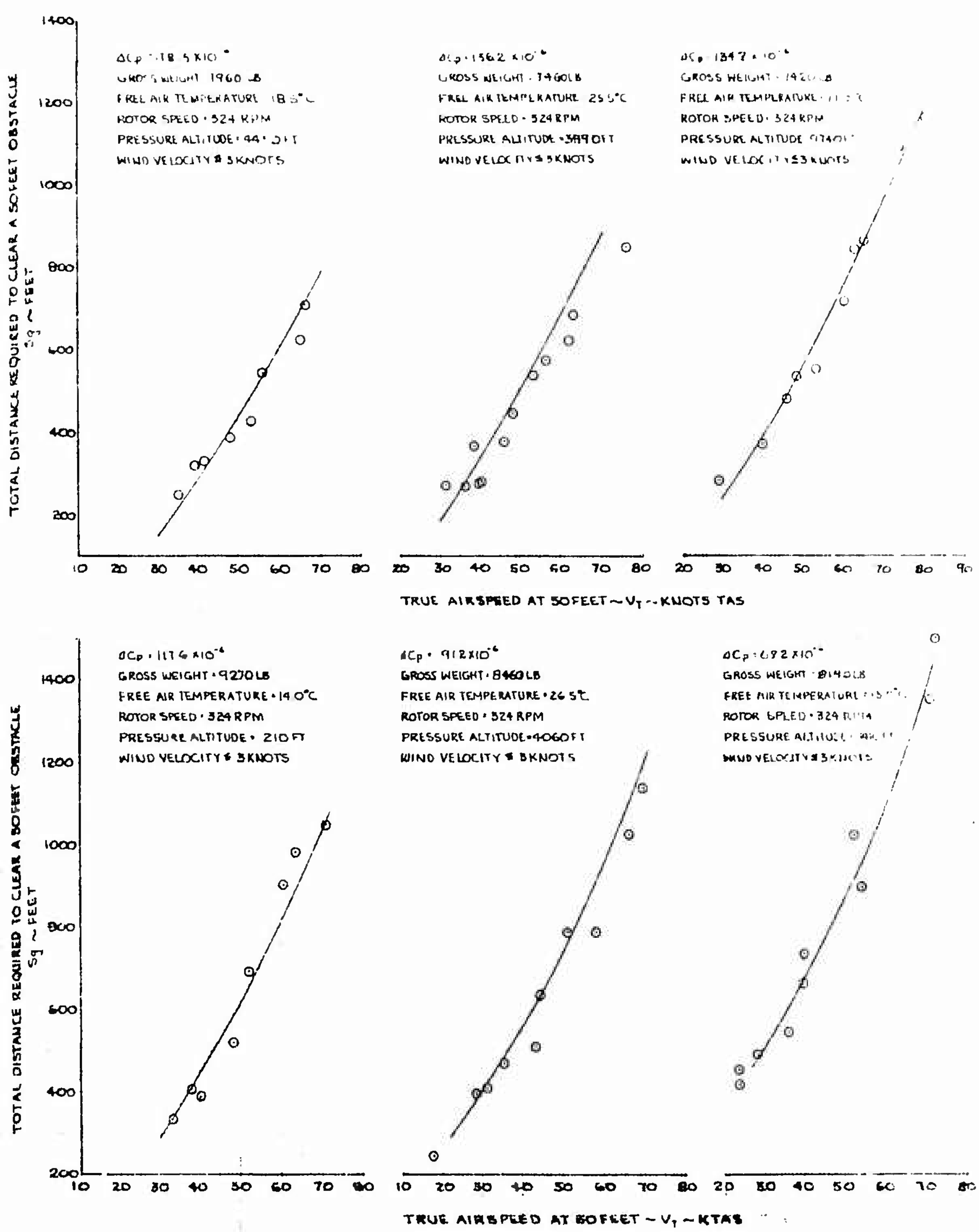


FIGURE NO 29

TAKE OFF PERFORMANCE

AH-1G USA S/N 615247

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

ROTOR SPEED = 324 RPM

CENTER OF GRAVITY = 195 INCHES (MID)

TECHNIQUE: LEVEL FLIGHT ACCELERATION FROM A 3 FEET HOVER

NOTES 1. $\Delta C_p = C_p$ AT 50 FEET - C_p TO HOVER AT 3 FEET

2. C_p AT 50 FEET - BASED ON ENGINE POWER AVAILABLE UNDER TEST DAY ATMOSPHERIC CONDITIONS

3. C_p TO HOVER AT 3 FEET - BASED ON ENGINE POWER REQUIRED TO HOVER AT 3 FEET UNDER TAKE-OFF ATMOSPHERIC CONDITIONS

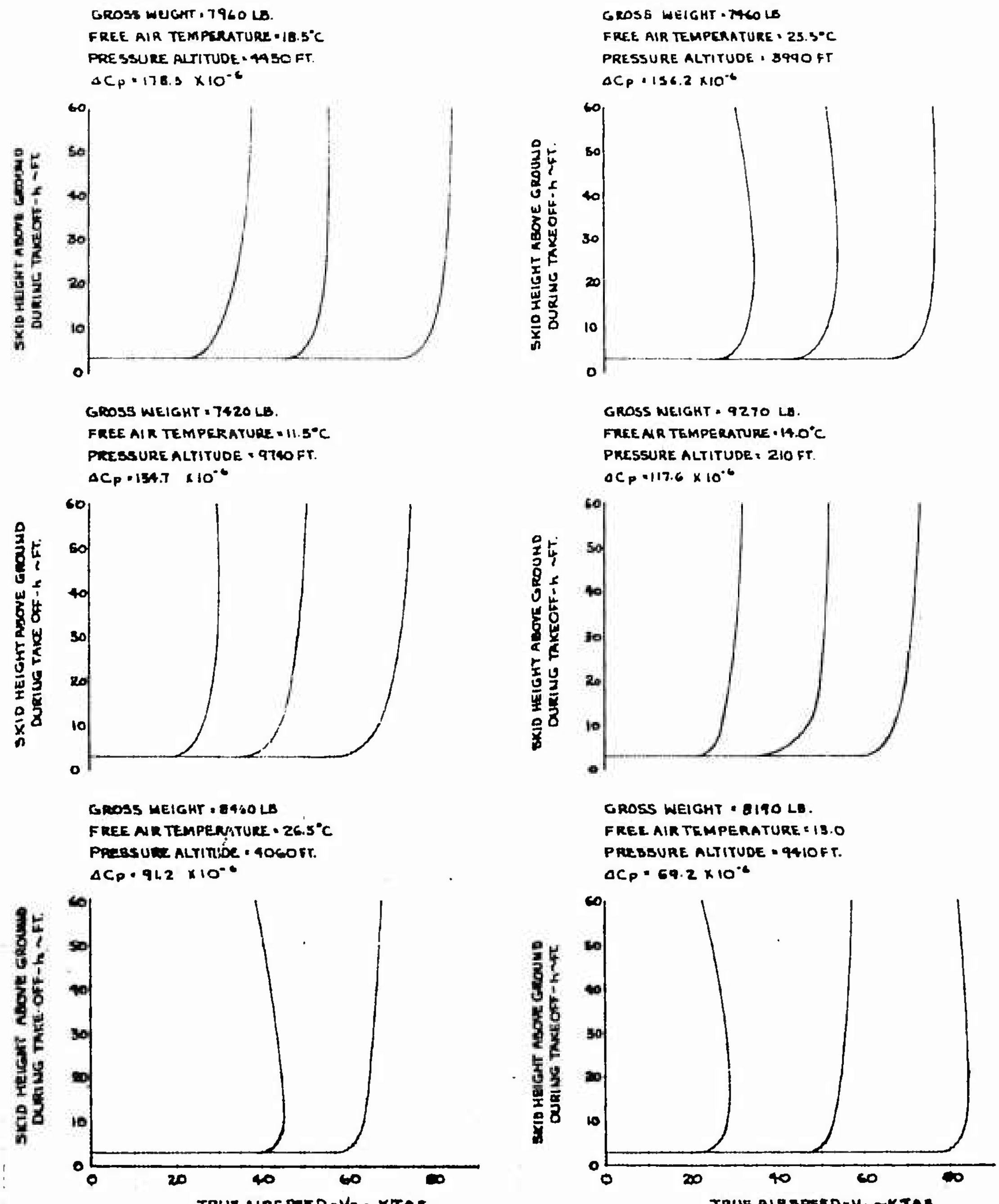


FIGURE No 30

TAKEOFF TIME HISTORY

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

TECHNIQUE LEVEL FLIGHT ACCELERATION FROM A 3 FEET HOVER

ALTITUDE AT A HOVER - H_p ~ FT	GROSS WT ~ LB	LONG C G ~ IN	ROTOR SPEED ~ RPM	THRUST COEFFICIENT
11140	8146	196.0	324	0.005680

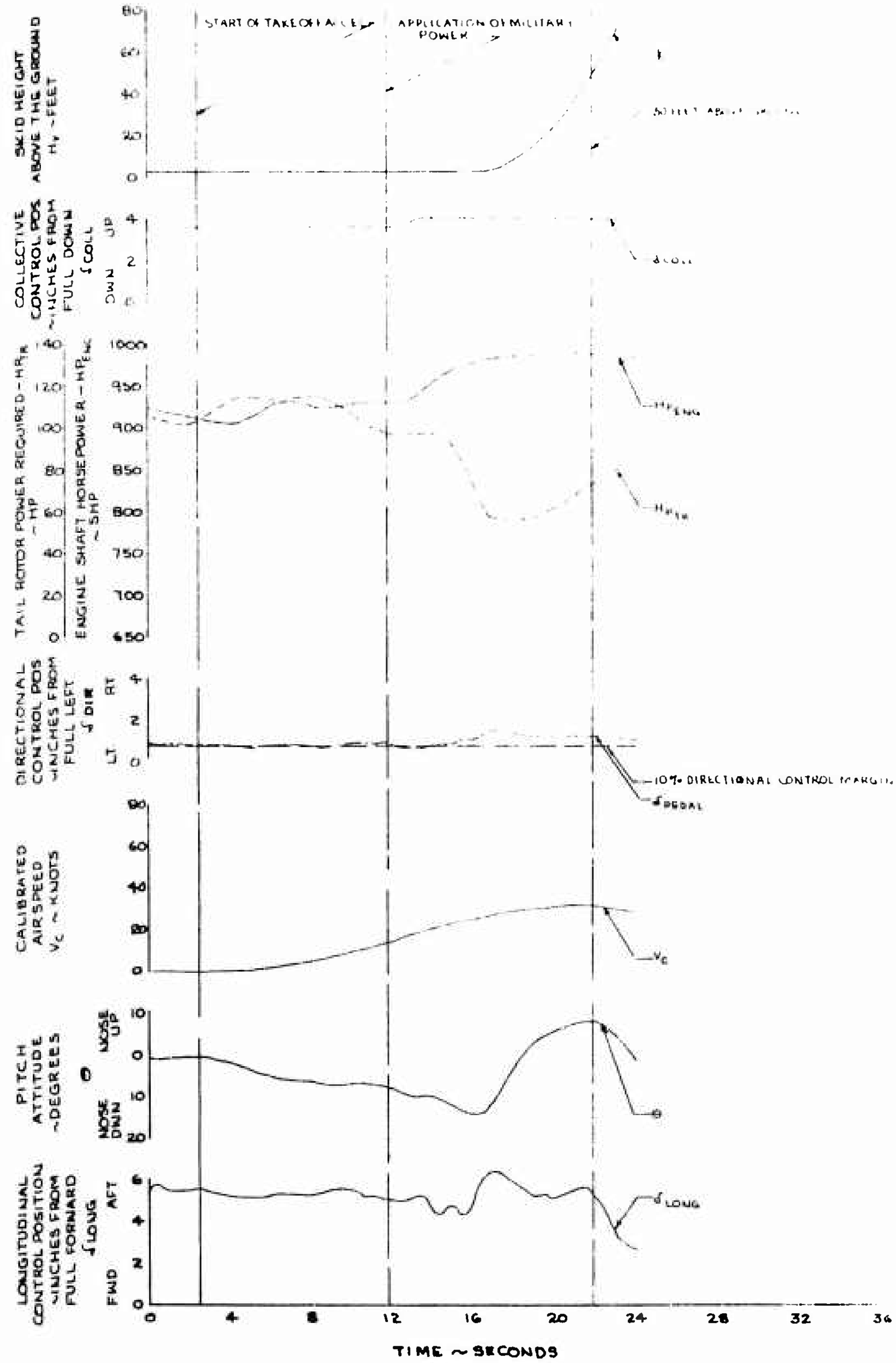


FIGURE No. 31

TAKE OFF TIME HISTORY

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

TECHNIQUE: LEVEL FLIGHT ACCELERATION FROM A 3 FEET HOVER

ALTITUDE AT A HOVER ~ FT	GROSS WT ~ LB	LONG C G ~ IN	ROTOR SPEED ~ RPM	THRUST COEFFICIENT
11140	8160	196 0	324	0 005688

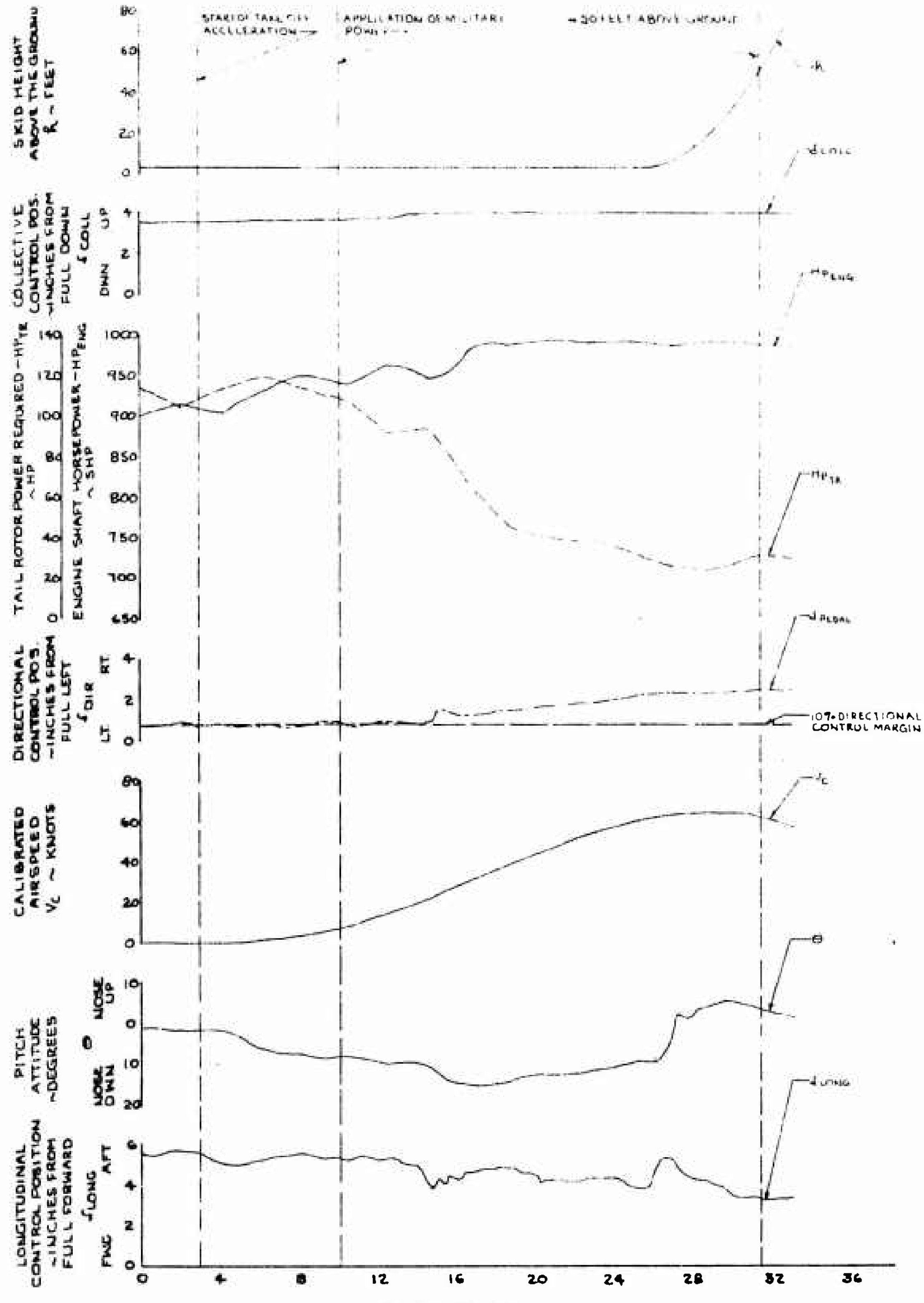


FIGURE NO. 32

CLIMB PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

OUTBOARD ALTERNATE CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CONTRACT GUARANTEE COMPLIANCE CHECK

ENGINE PARTICLE SEPARATOR NOT INSTALLED

NOTES: 1. CLIMB START GROSS WEIGHT = 8000 LB.

2. ROTOR SPEED = 324 RPM

3. C.G. STATION = 192.0 INCHES (FORWARD)

4. STANDARD DAY

5. ENGINE POWER AVAILABLE OBTAINED FROM FIGURE NO. 118 APP. VII

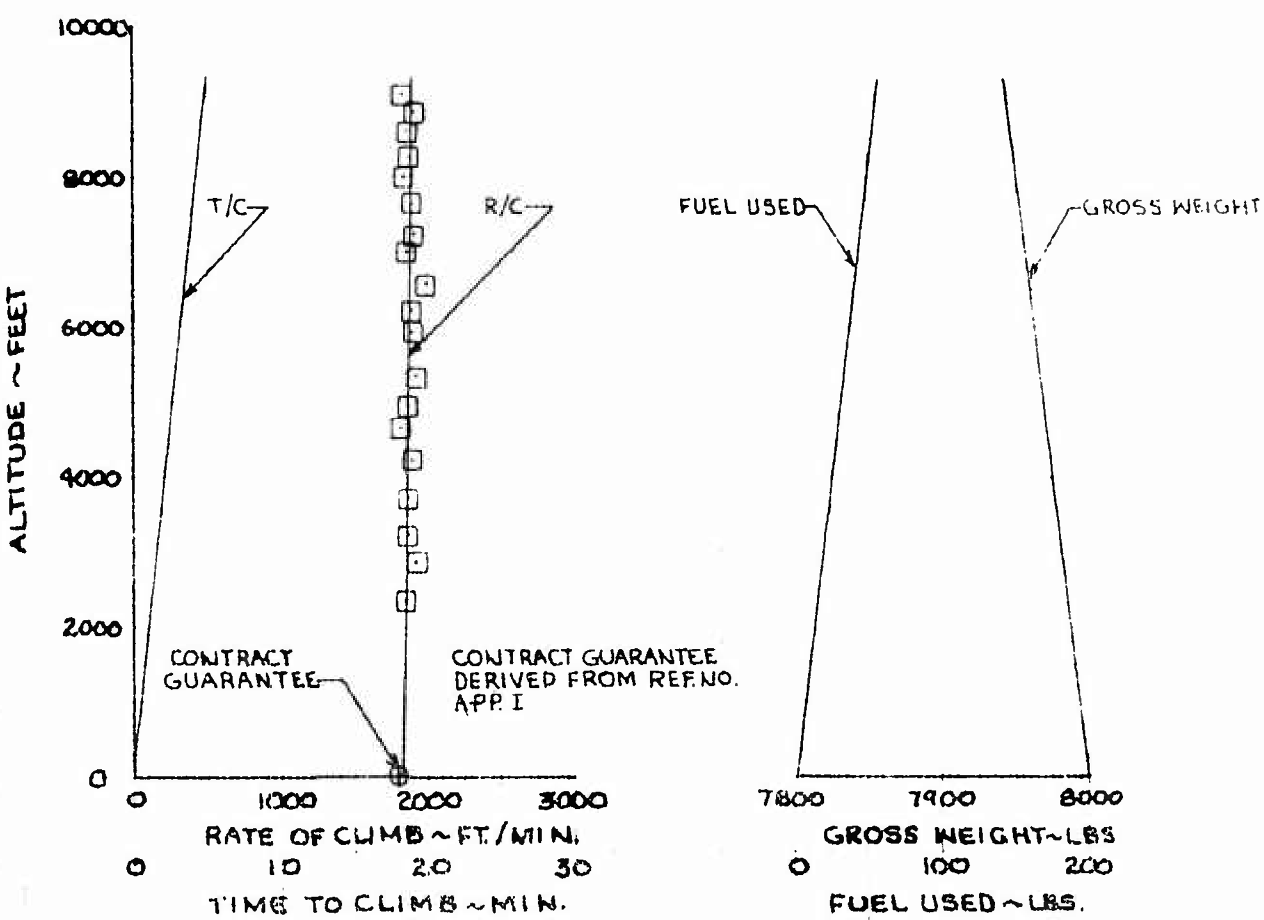


FIGURE No. 32 CONTINUED

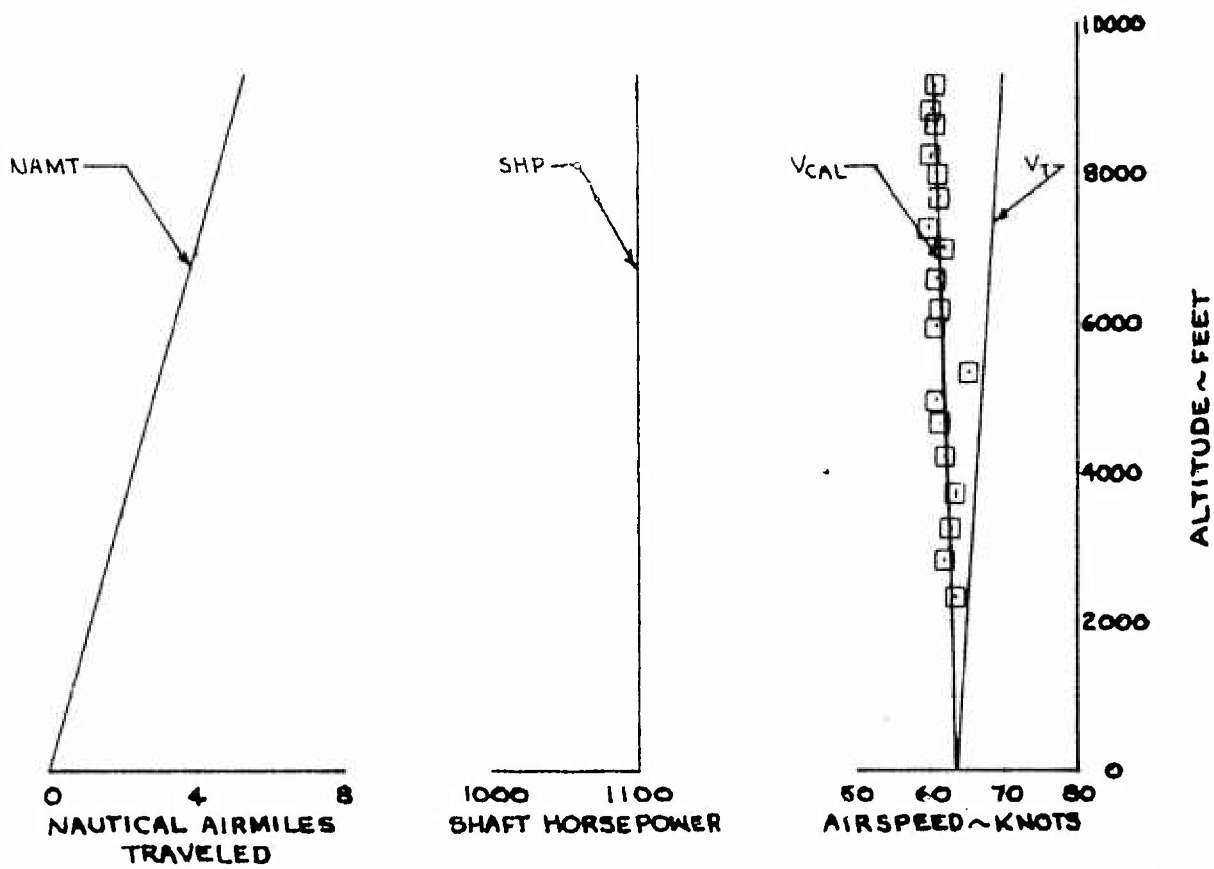


FIGURE No 33

CLIMB PERFORMANCE

AH-1G USA S/N 615247

CLEAN CONFIGURATION

ENGINE PARTICLE SEPARATOR INSTALLED

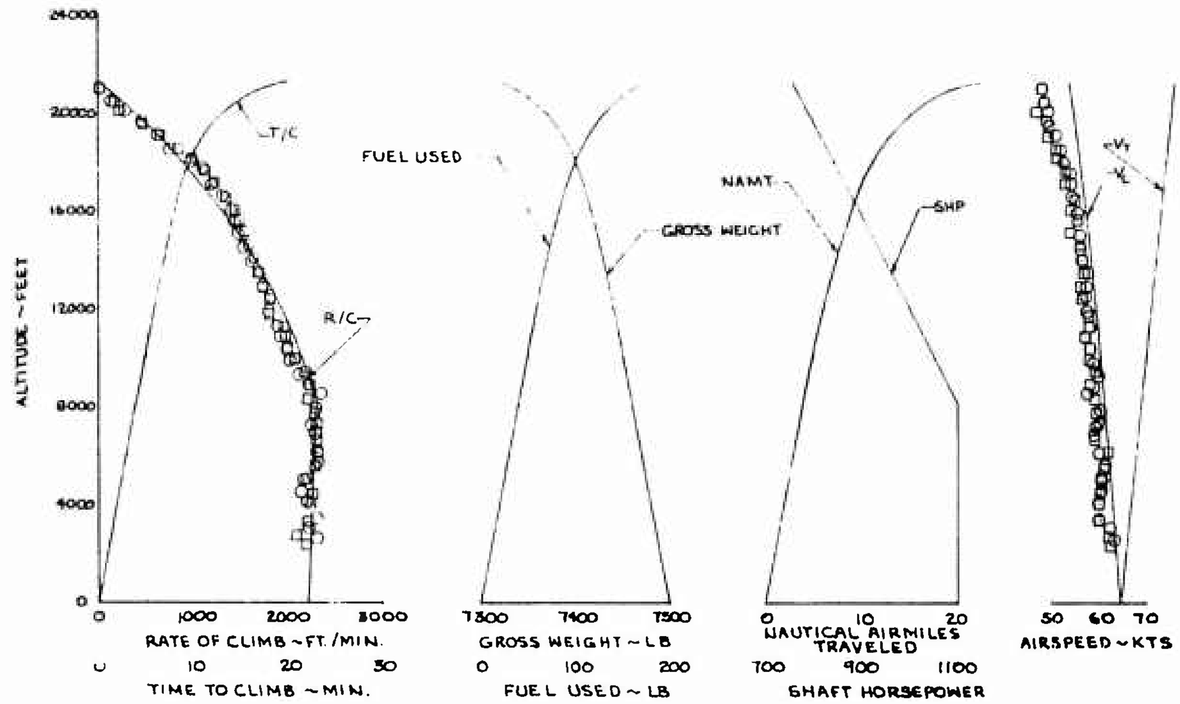
NOTES 1 ENGINE POWER AVAILABLE OBTAINED FROM FIGURE NO 116 APP VII

2 ROTOR SPEED = 324 RPM

3 STANDARD DAY

CLIMB START GROSS WEIGHT = 7500 LB

CENTER OF GRAVITY = 191.2 IN (FWD)



CLIMB START GROSS WEIGHT = 8500 LB.

CENTER OF GRAVITY = 191.4 IN. (FWD.)

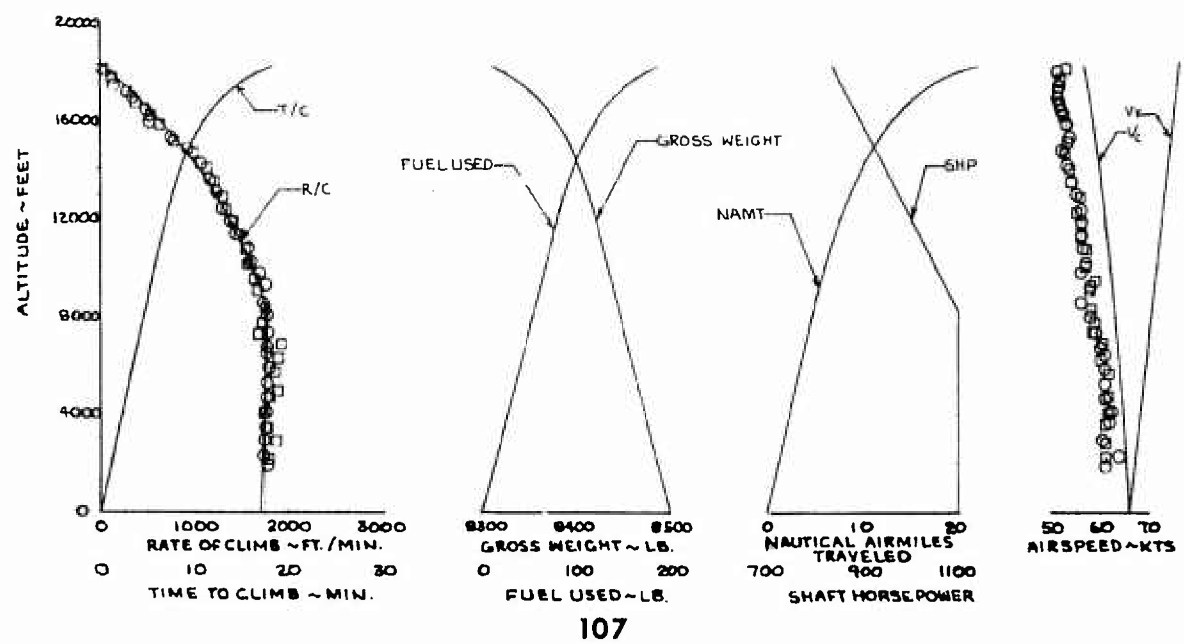


FIGURE NO 34

CLIMB PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

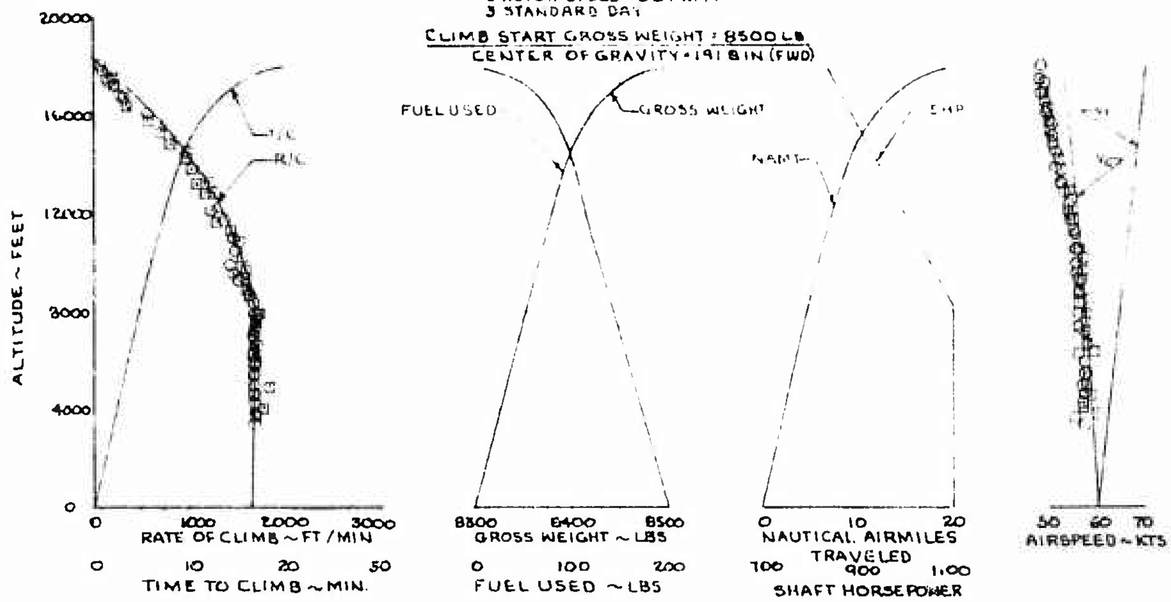
HVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES 1 ENGINE POWER AVAILABLE OBTAINED FROM FIGURE NO 116 APP'X D

2 ROTOR SPEED = 324 RPM

3 STANDARD DAY



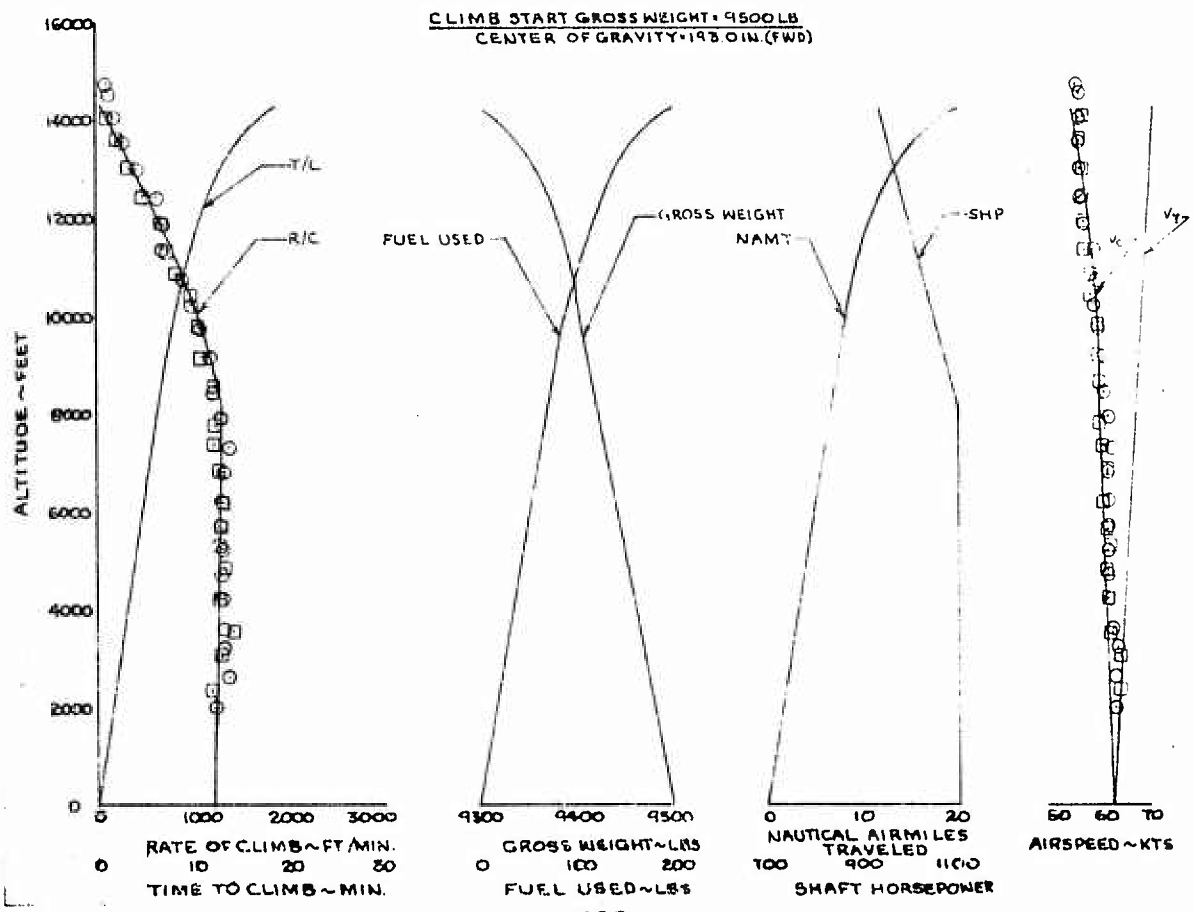
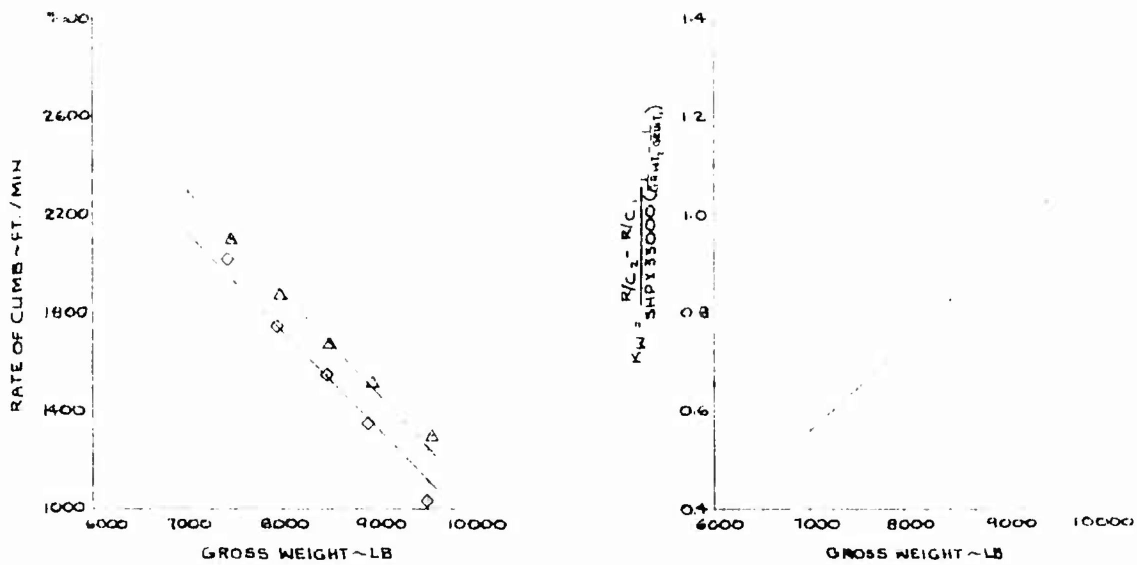


FIGURE NO 35
VARIATION IN RATE OF CLIMB AS A FUNCTION
OF GROSS WEIGHT AND
ENGINE SHAFT HORSEPOWER
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYM	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	ENGINE SHP	CONFIGURATION	LONG C.G. ~INCHES
	5000	324	1100	HVY HOG	FWD
	10000	324	1050	HVY HOG	FWD



SYM	DENSITY ALTITUDE ~FT	GROSS WEIGHT ~LB	ROTOR SPEED ~RPM	CONFIGURATION	LONG C.G. ~INCHES
○	5000	8500	324	HVY HOG	FWD
□	5000	8500	324	CLEAN	FWD

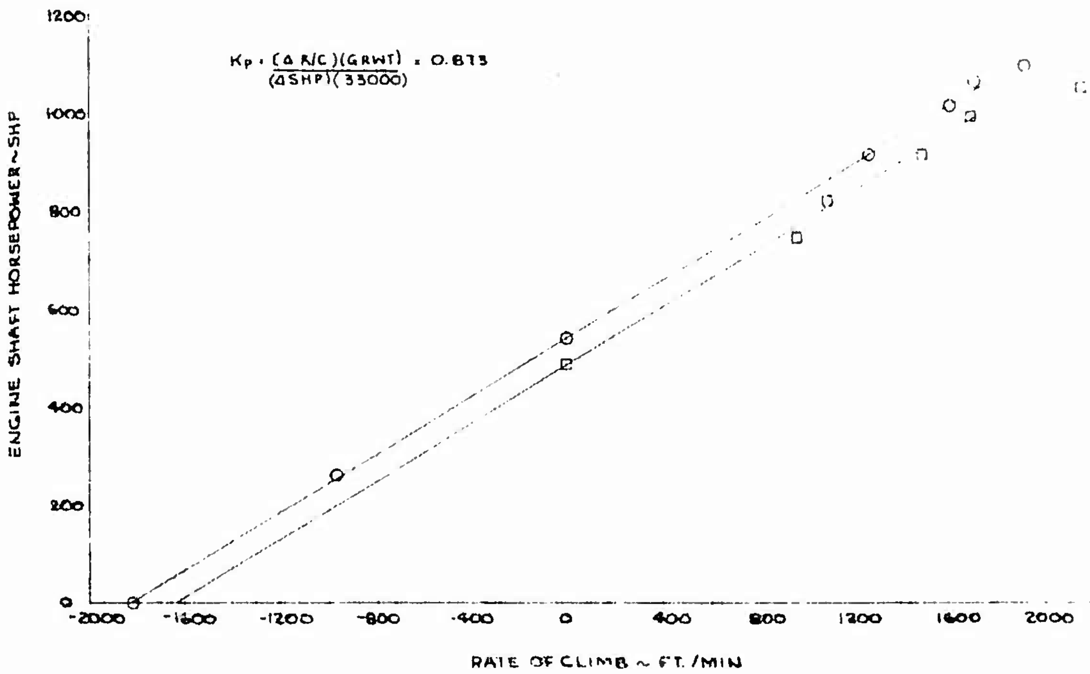


FIGURE NO. 36

NON-DIMENSIONAL ROTOR TIP SPEED RATIO FOR MAXIMUM CLIMB PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

CENTER OF GRAVITY - FORWARD

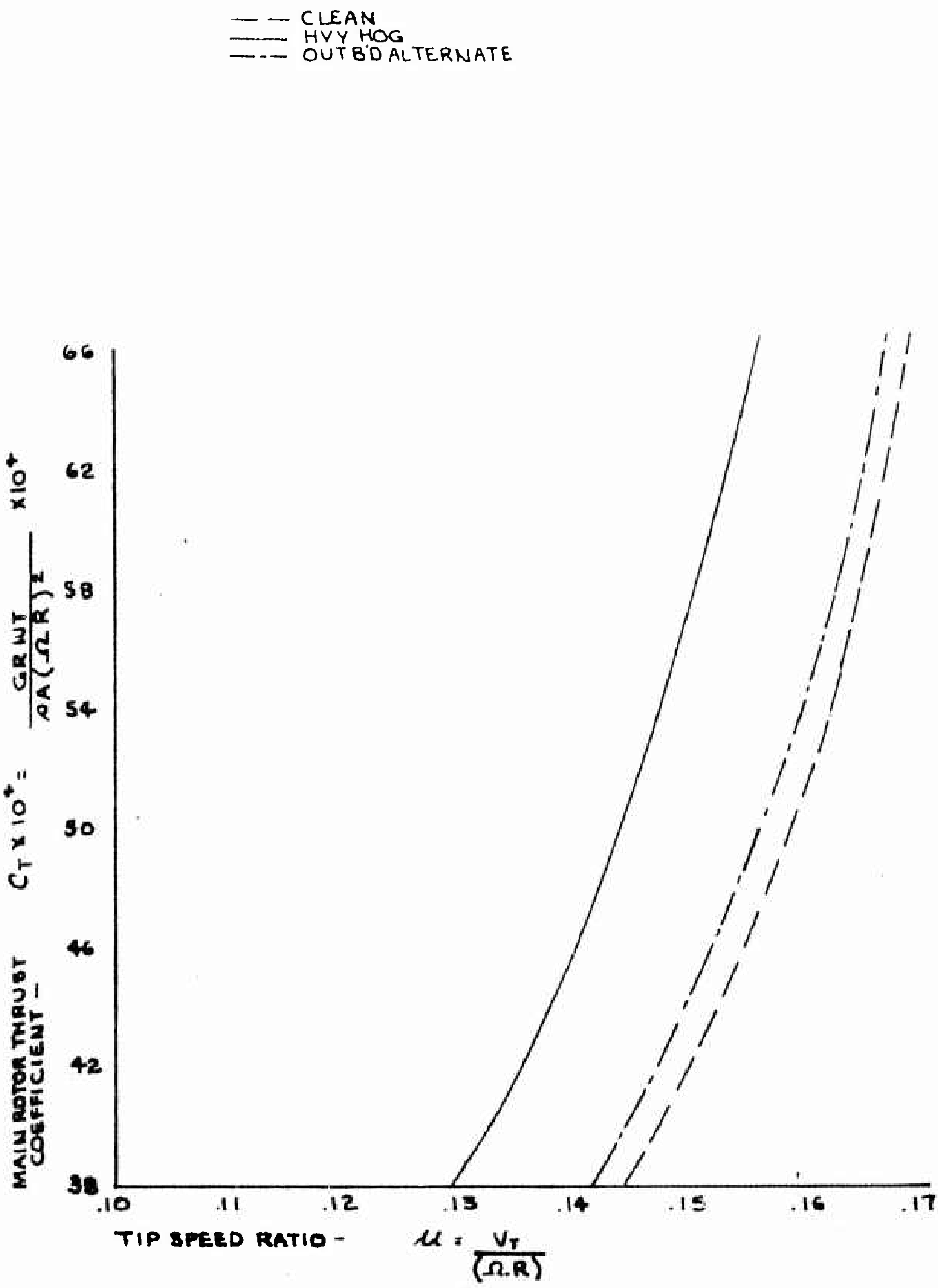


FIGURE NO 37

NON DIMENSIONAL MINIMUM POWER REQUIRED

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

CENTER OF GRAVITY - FORWARD

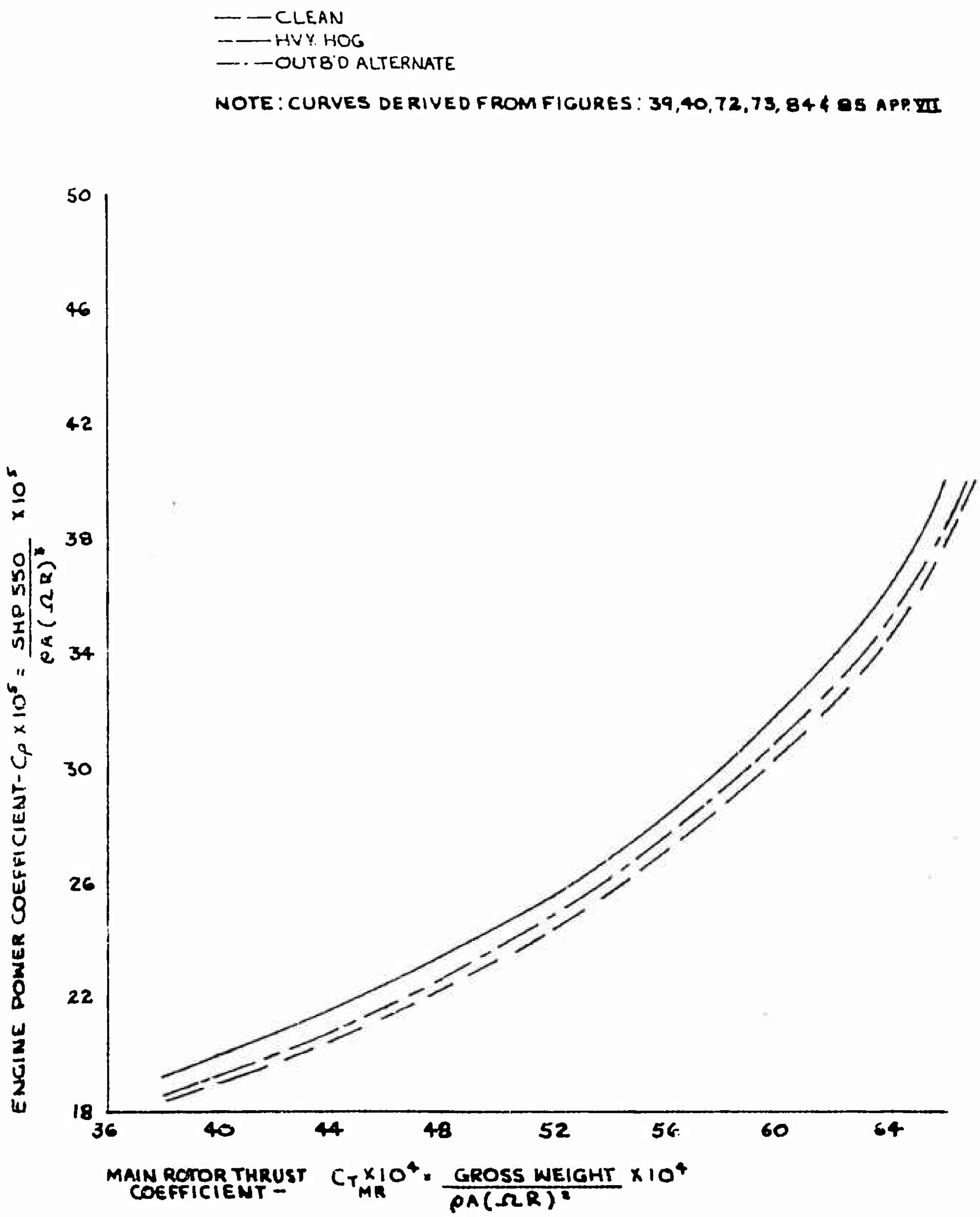


FIGURE NO 38

CHANGE IN EQUIVALENT FLAT PLATE AREA
DUE TO WING ARMAMENT CONFIGURATION CHANGES

AH-1G USA S/N 615247

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 5000 FT.
GROSS WEIGHT = 8500 LB
$C_T = 49.00 \times 10^{-4}$
C.G. LOCATION = FORWARD

NOTES: 1. ALL ROCKET PODS UNFAIRED
2. CURVES DERIVED FROM FIGURES: 40, 57, 65, 69, 73, 76, 79, & 85 APP. VII

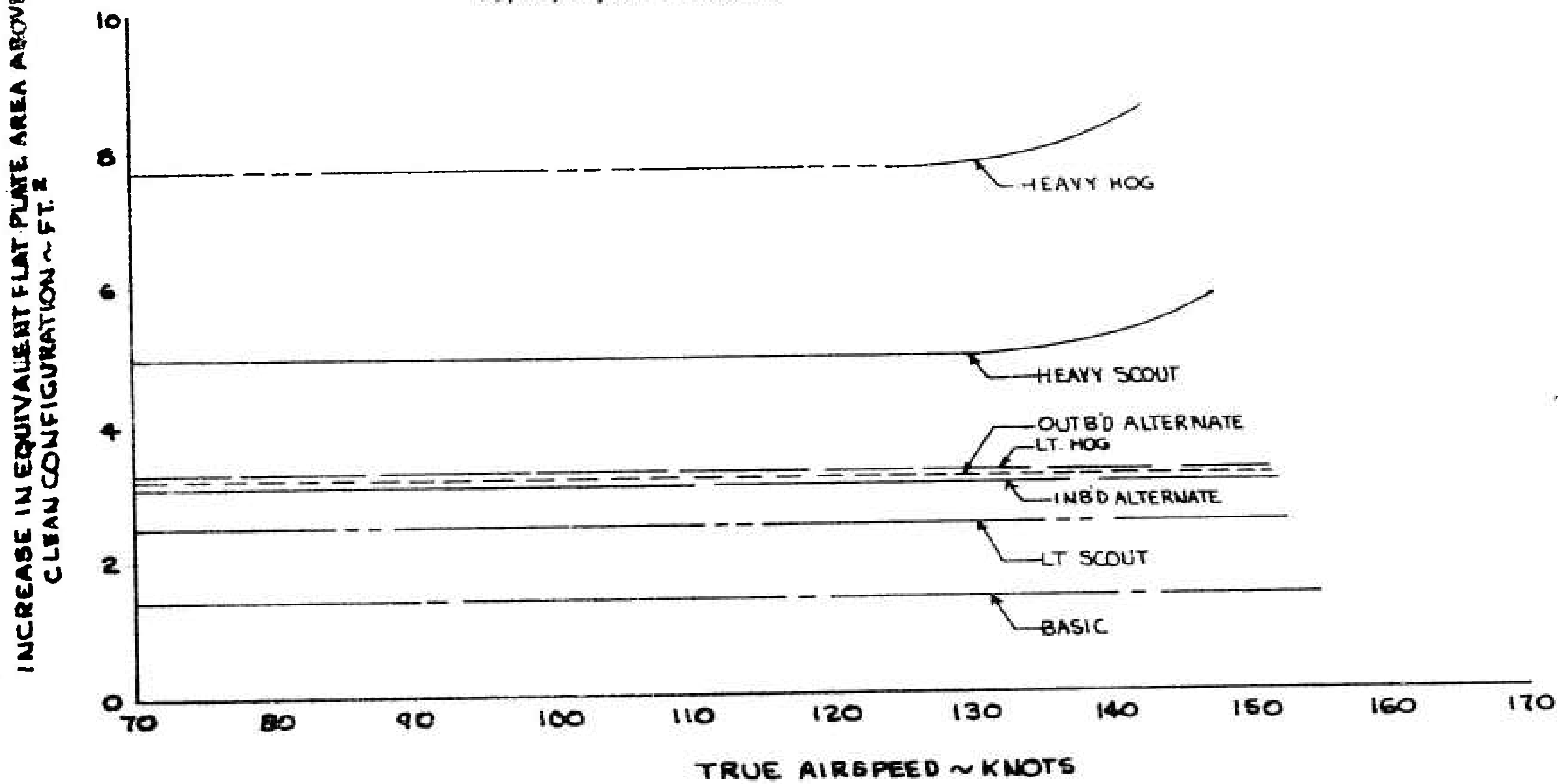


FIGURE No. 39

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

CLEAN CONFIGURATION

CENTER OF GRAVITY - FORWARD

ROTOR SPEED - 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 41 THROUGH 46 APP VII

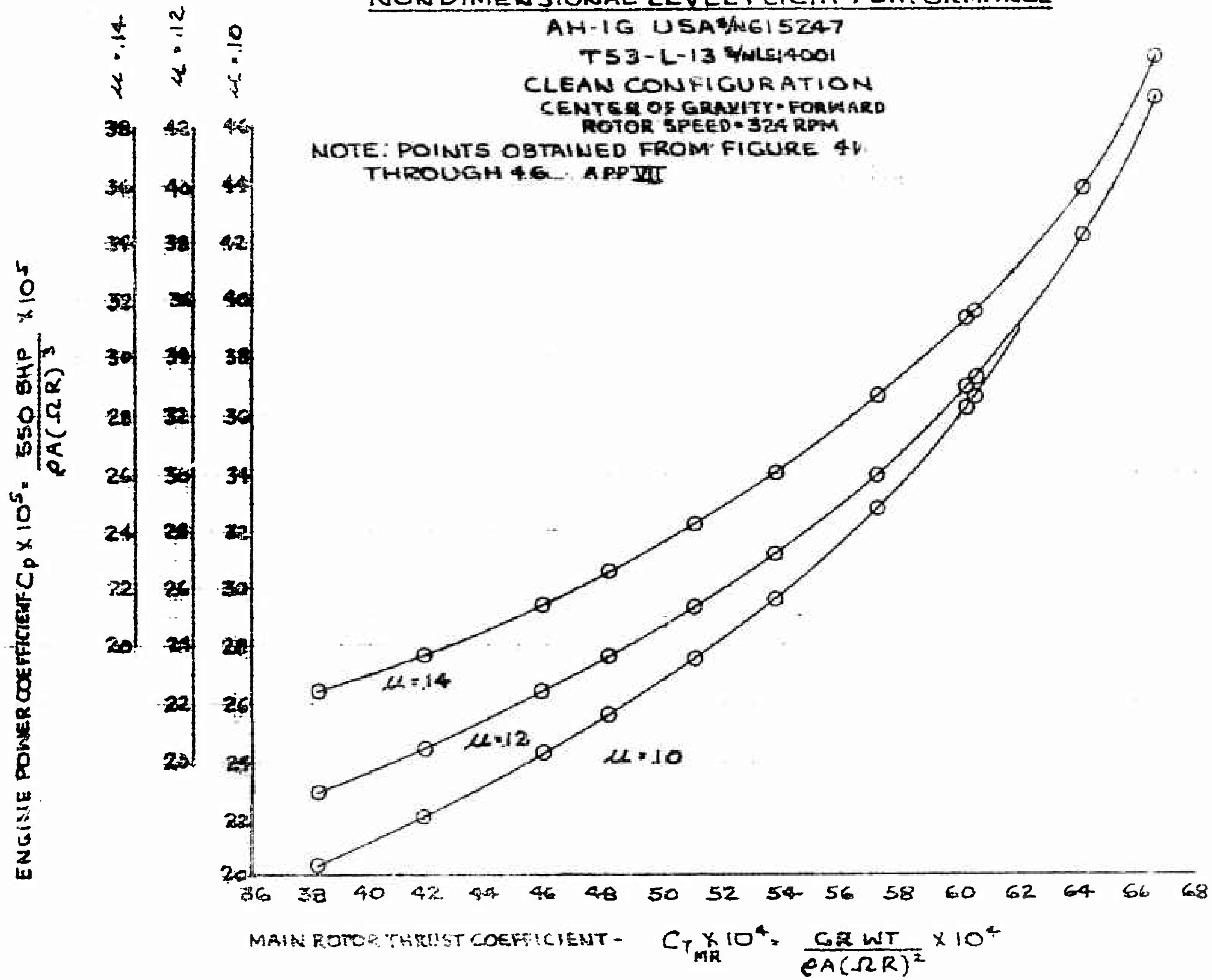


FIGURE No 40

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

CLEAN CONFIGURATION

CENTER OF GRAVITY - FORWARD

ROTOR SPEED = 324 RPM

NOTE POINTS OBTAINED FROM FIGURE 41 THROUGH 46 APP VII

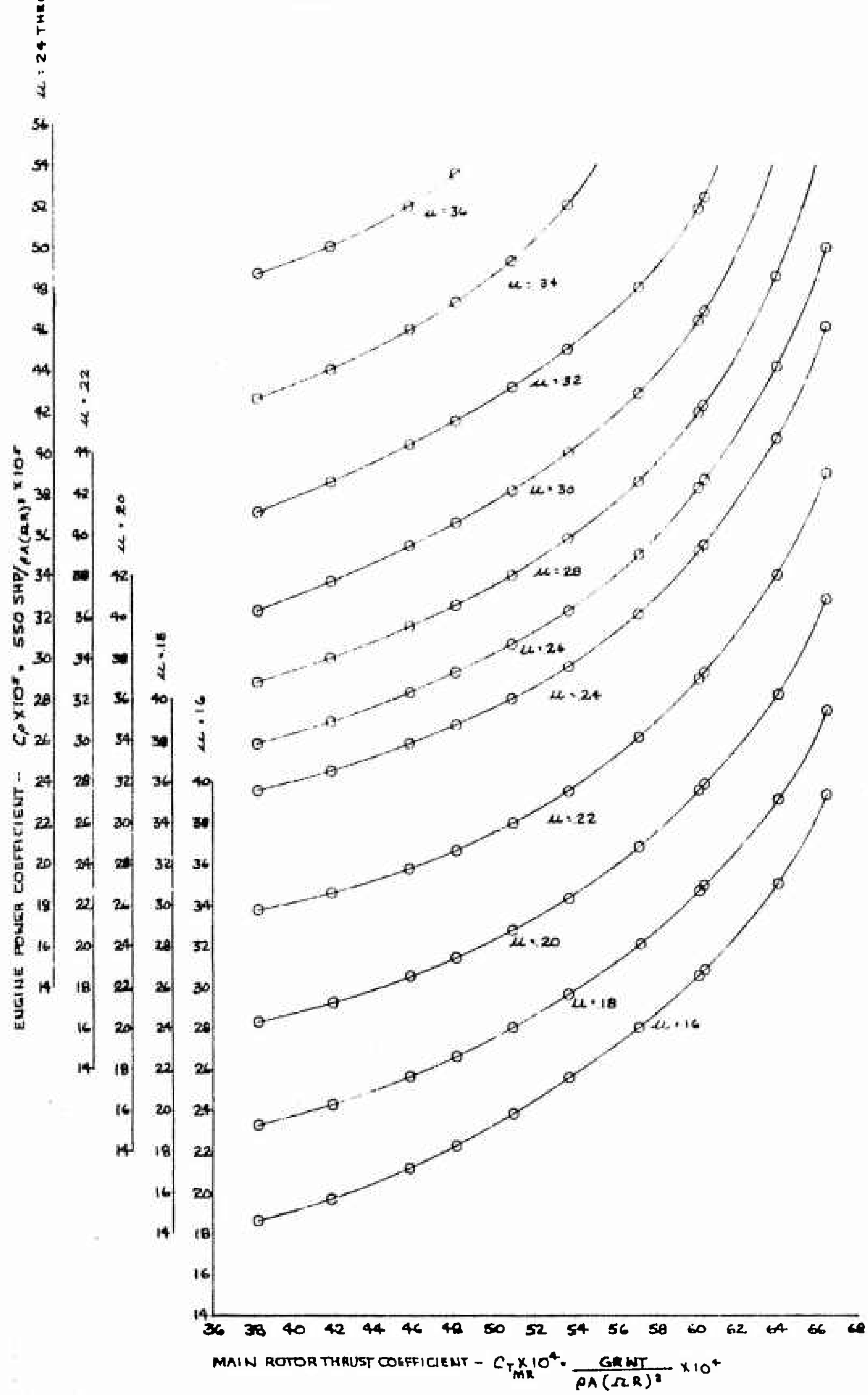


FIGURE NO 41
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYMBOL	AVG ALTITUDE H_D ~ FEET	AVG GROSS WEIGHT ~ LB	AVG LONG C G ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	2150	7250	191 0 (FWD)	0 003823	324	CLEAN
□	4670	7350	191 2 (FWD)	0 004181	324.5	CLEAN

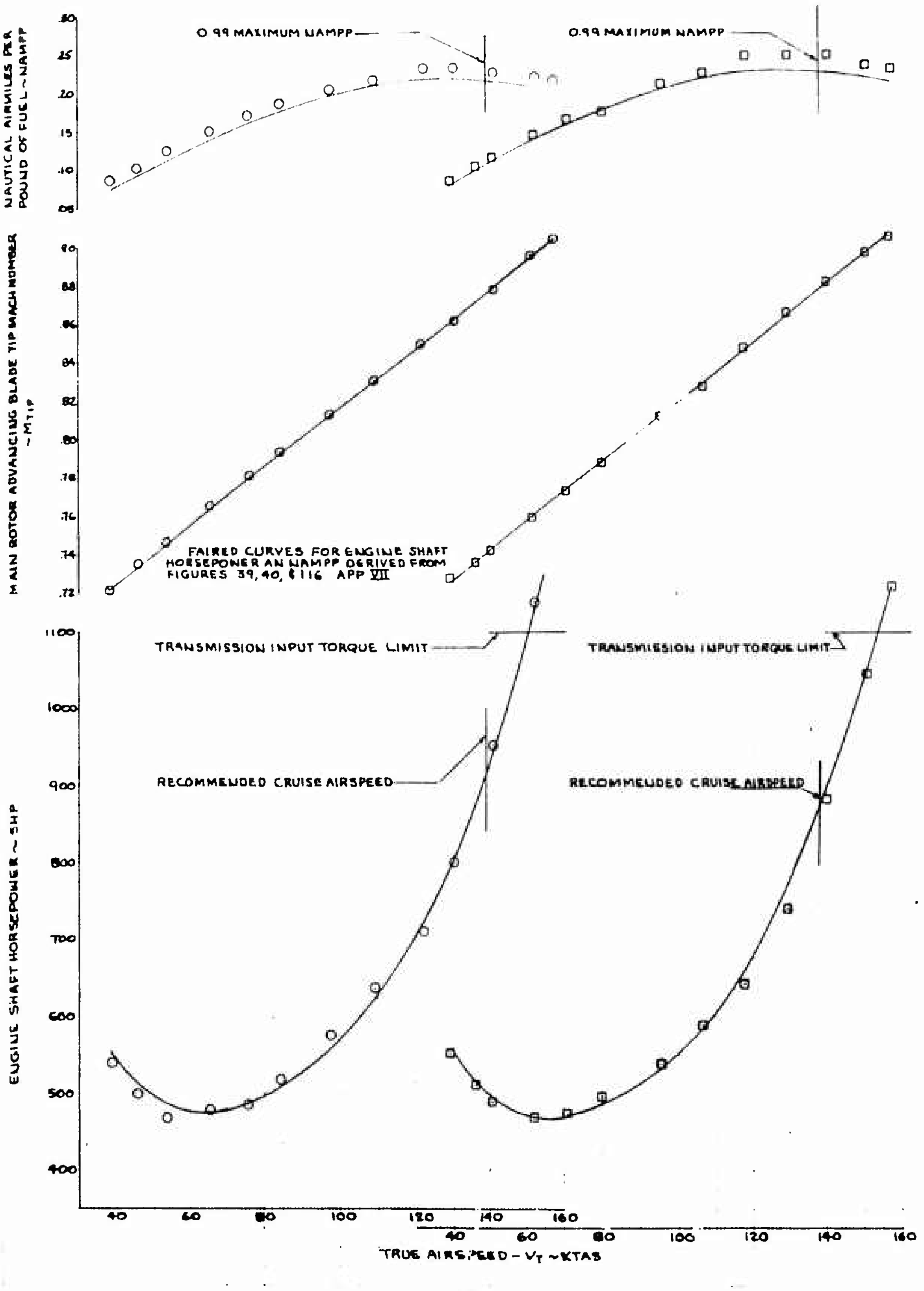


FIGURE NO 42

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	AVG ALTITUDE H_p ~ FEET	AVG GROSS WEIGHT ~ LB.	AVG LONG C.G ~ IN	AVG THRUST COEFF. ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
	4360	8150	192.0(FWD)	0.004584	325	CLEAN
	4220	8350	192.5(FWD)	0.004814	324	CLEAN

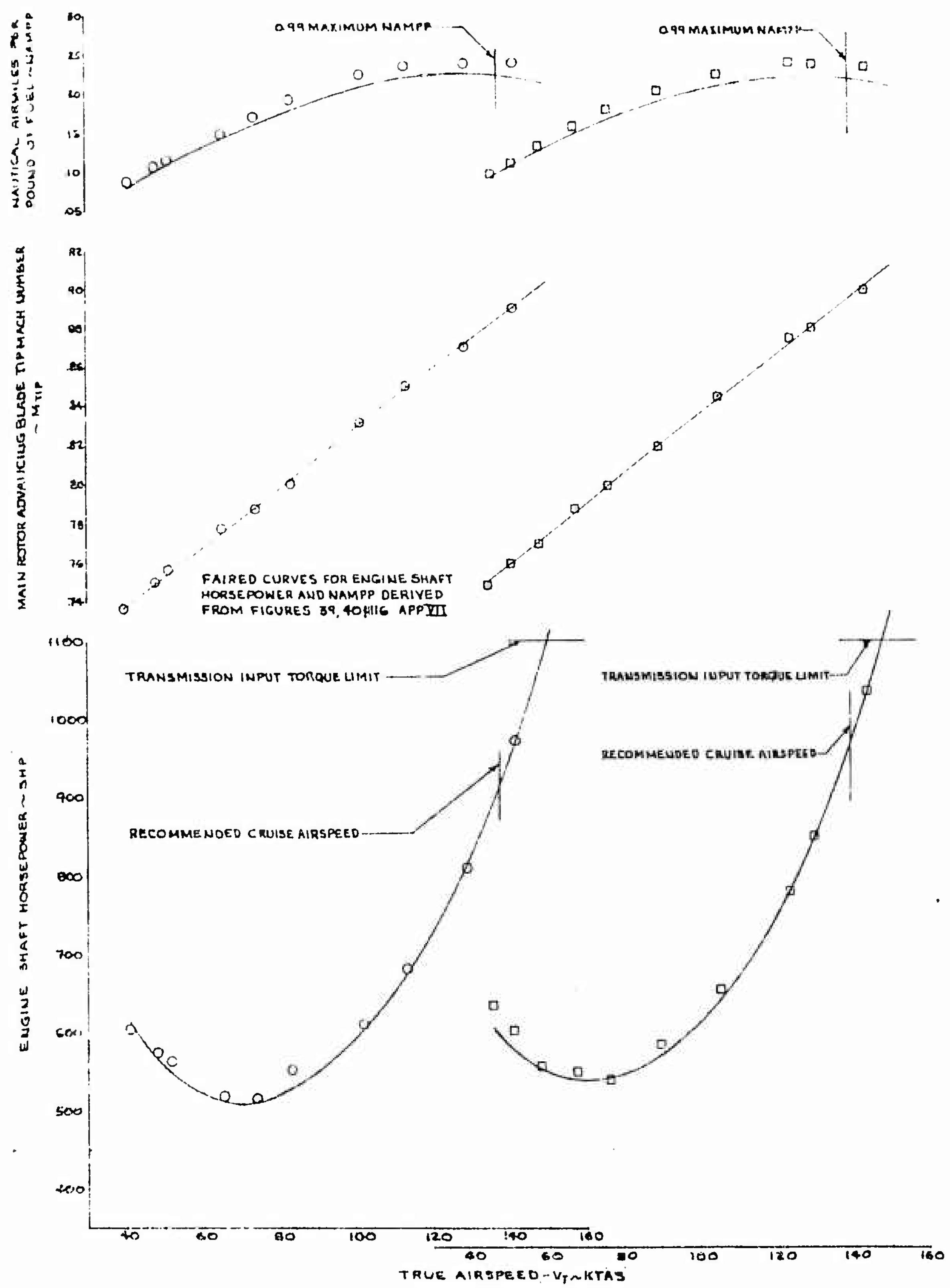


FIGURE NO 43

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615241

T53-L-13 S/N LE14001

SYMBOL	AVG ALTITUDE H_D ~ FEET	AVG GROSS WEIGHT ~ LB	AVG LONG C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
	9560	7160	197.1 (FWD)	0.005099	324.5	CLEAN
	4840	8030	191.9 (FWD)	0.005312	324	CLEAN

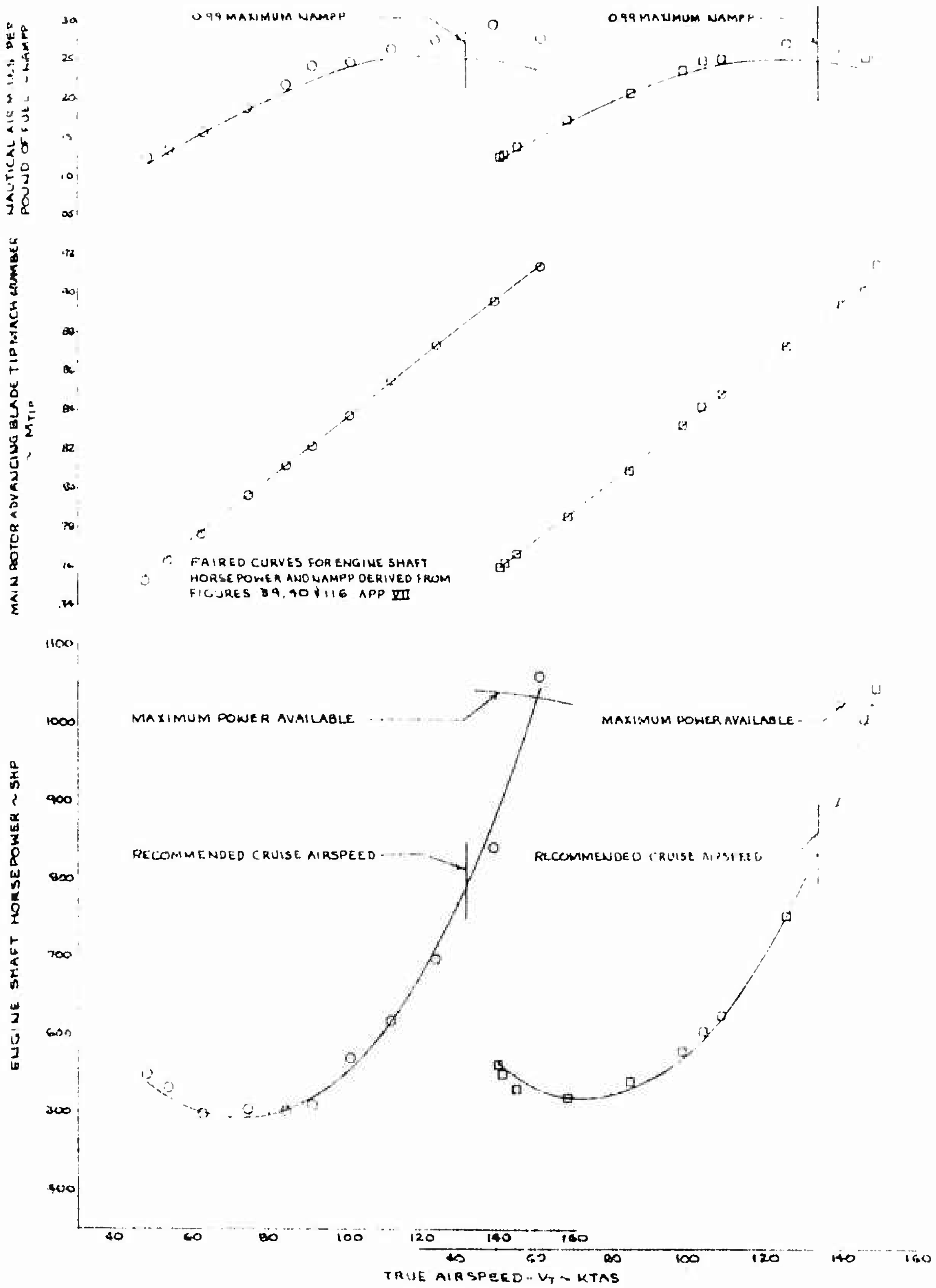


FIGURE NO 44
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYMBOL	AVG. ALTITUDE H_D ~ FEET	AVG. GROSS WEIGHT ~ LB	AVG. LONG C.G ~ IN	AVG. THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	9740	8540	191.4 (FWD)	0.005714	323.5	CLEAN
□	9410	9120	192.5 (FWD)	0.006020	324	CLEAN

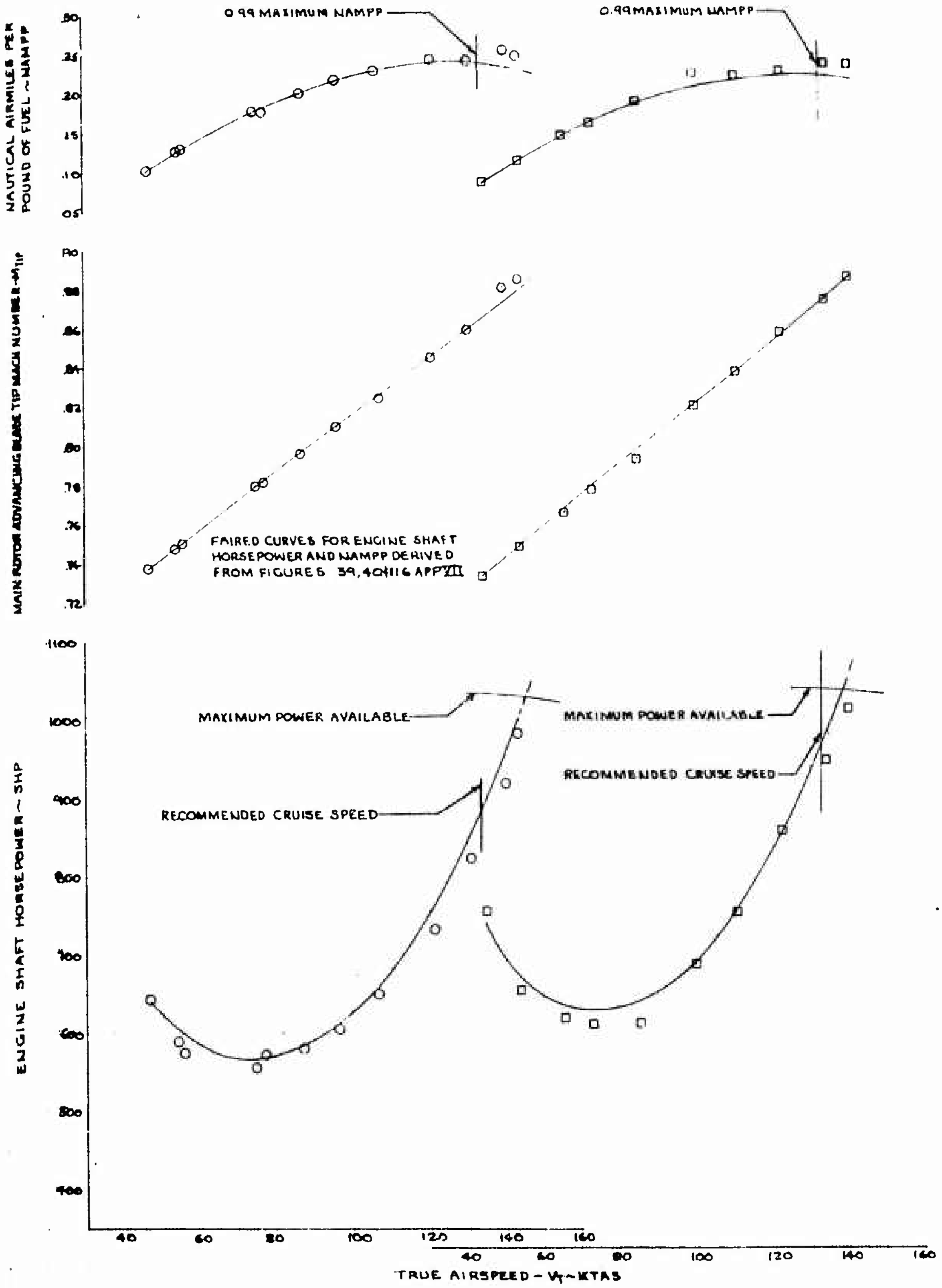


FIGURE NO 45

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	AVG ALTITUDE H_D ~FEET	AVG GROSS WEIGHT ~LB	AVG LONG C G ~IN.	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~RPM	ARMAMENT CONFIG.
○	15260	7600	191.7 (FWD)	0.006049	324	CLEAN
□	14780	8170	191.0 (FWD)	0.006421	323.5	CLEAN

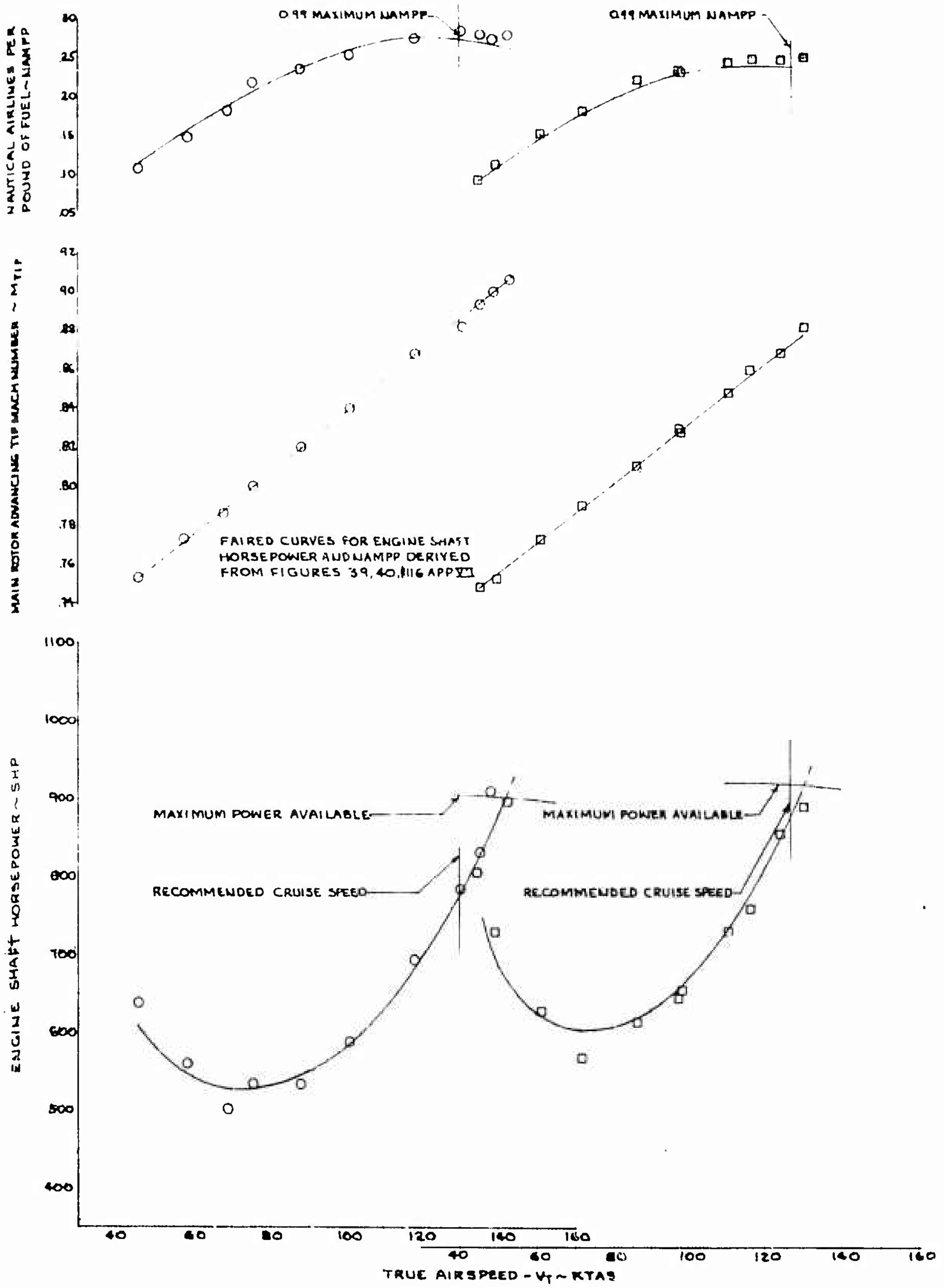


FIGURE NO 46

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	AVG ALTITUDE H_D ~ FEET	AVG GROSS WEIGHT ~ LB	AVG. LONG. C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
O	14270	8620	191 4 (FWD)	0 006664	323 5	CLEAN

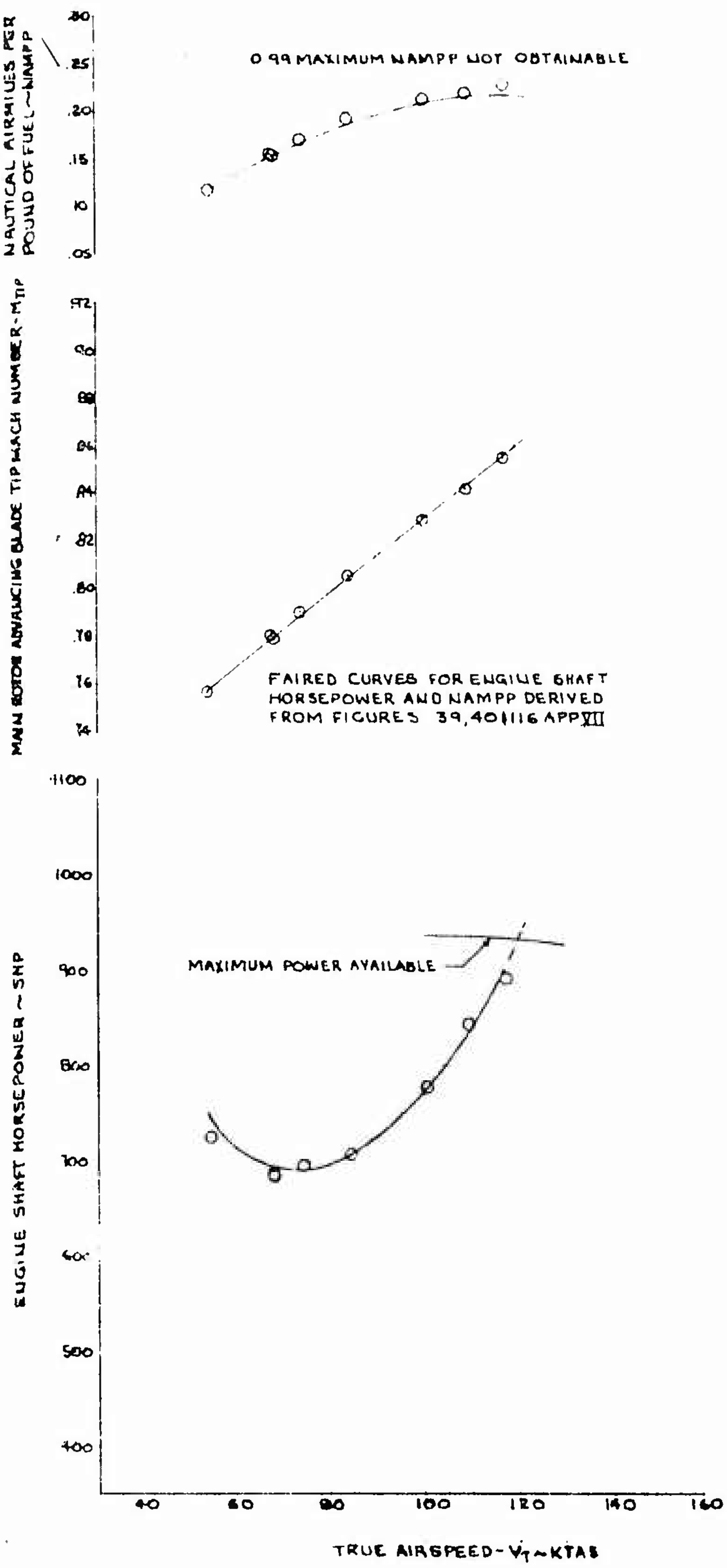


FIGURE NO. 47

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

CLEAN CONFIGURATION

LANDING GEAR CROSS TUBE FAIRINGS REMOVED

CENTER OF GRAVITY - FORWARD

ROTOR SPEED - 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 49 & 50

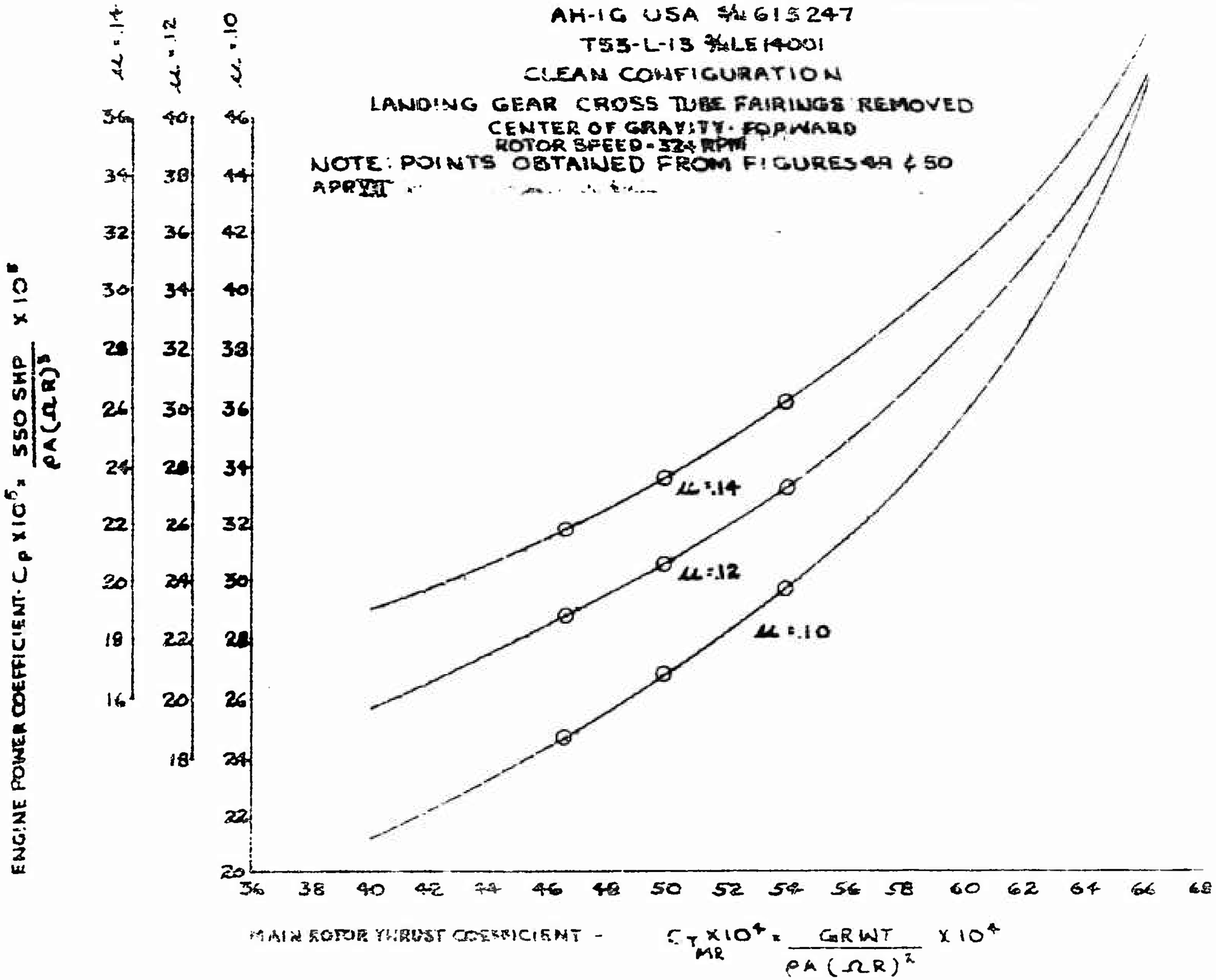


FIGURE NO 48

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

CLEAN CONFIGURATION

LANDING GEAR CROSS TUBE FAIRINGS REMOVED

CENTER OF GRAVITY = FORWARD

ROTOR SPEED = 324 RPM

NOTE POINTS OBTAINED FROM FIGURES 49 & 50 APP III

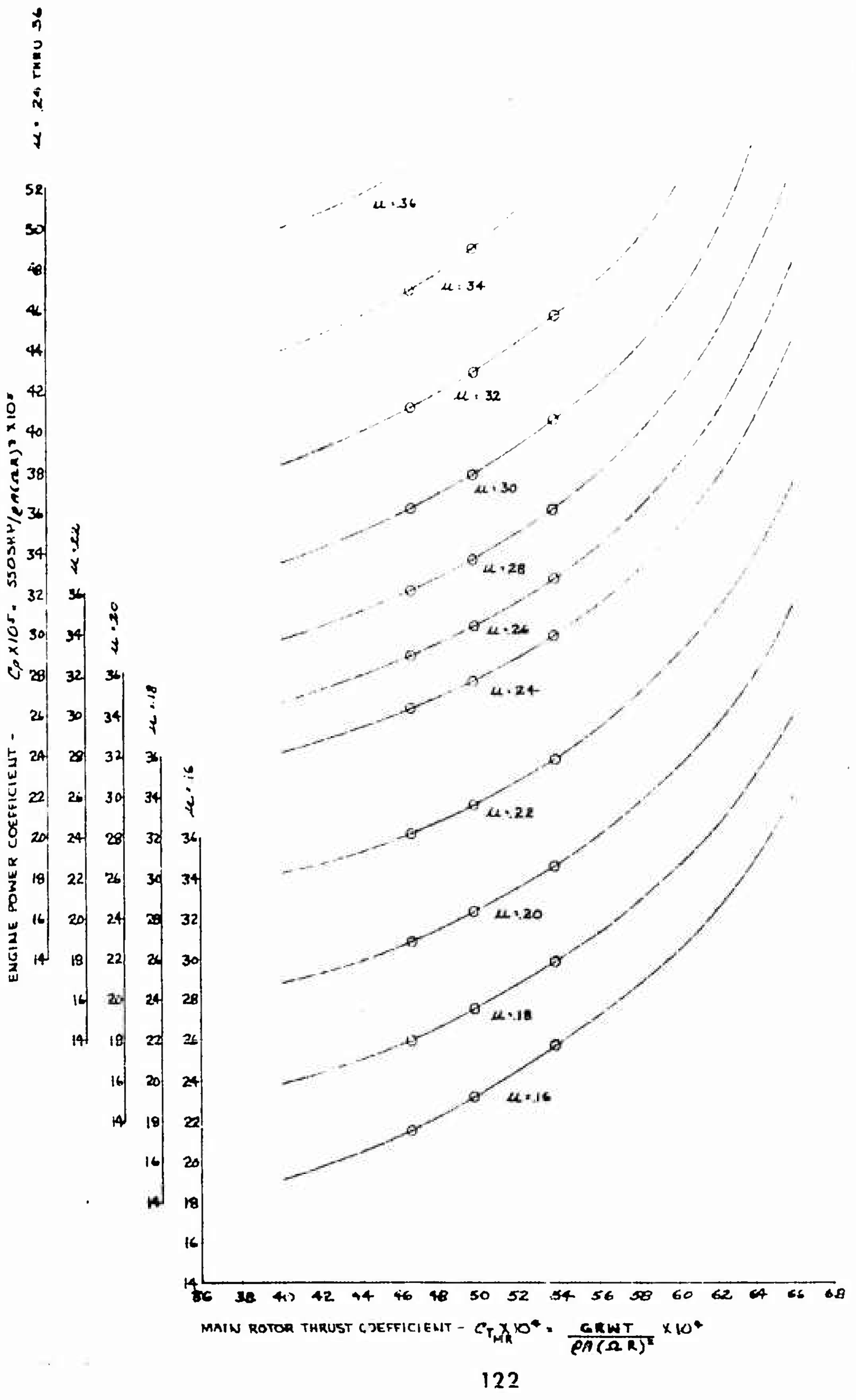


FIGURE No 49

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LS14001

SYMBOL	AVG. ALTITUDE H_D ~ FT.	AVG. GROSS WEIGHT ~ LB.	AVG. LONG. C.G. ~ IN.	AVG. THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	4080	8220	190 8 (FWD)	0 004650	322 5	CLEAN
□	4450	8760	191 4 (FWD)	0 004980	323 5	CLEAN

NOTE: LANDING GEAR CROSS TUBE FAIRINGS REMOVED

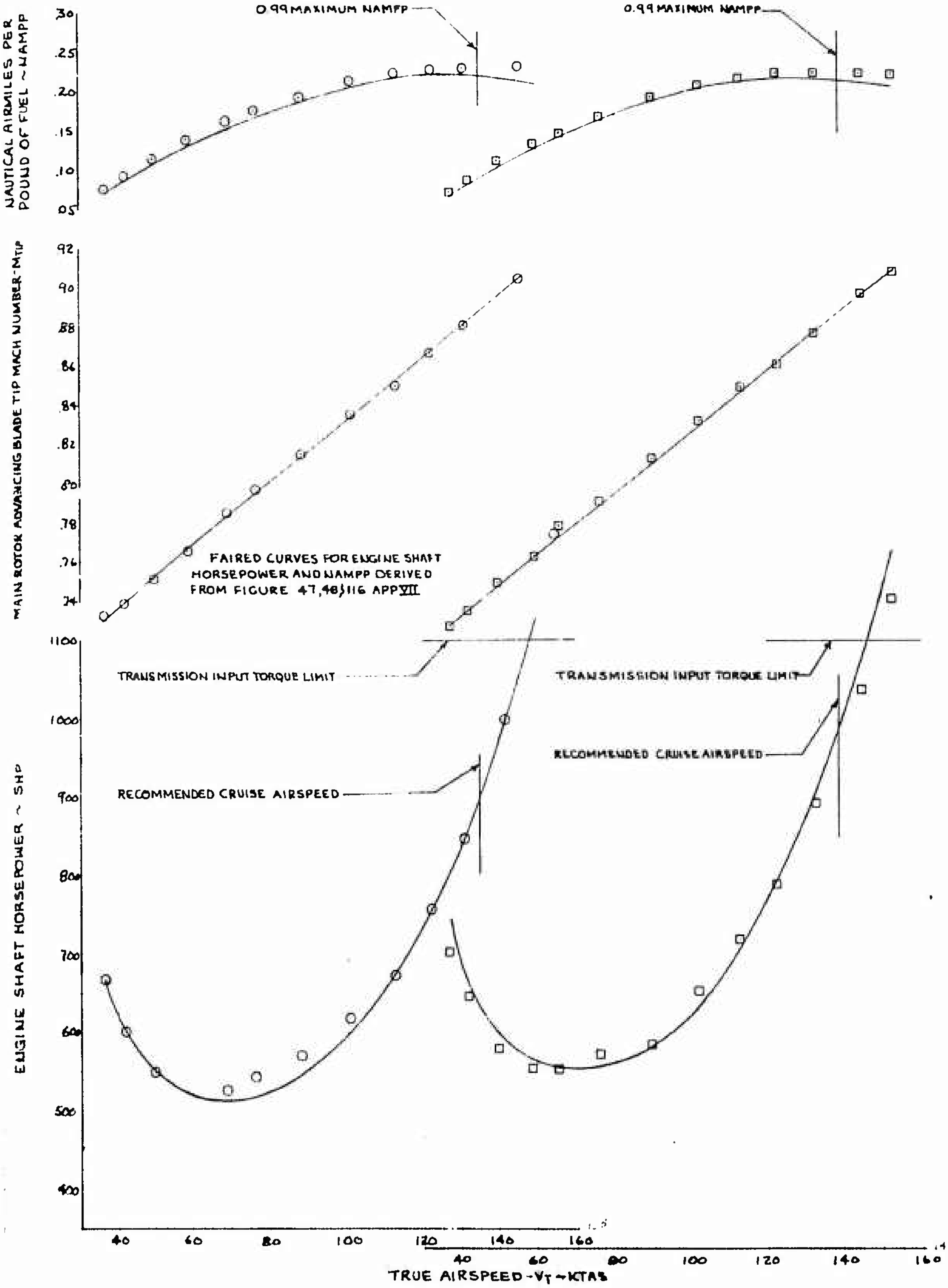


FIGURE NO. 50
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 66-15247
T53-L-13 S/N LE14001

SYMBOL	AVG. ALTITUDE H_D ~ FEET	AVG. GROSS WEIGHT ~ LB	AVG. LONG. C.G. ~ IN	AVG. THRUST COEFF. ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
O	9000	8170	190.9 (FWD)	0.005388	322	CLEAN

NOTE: LANDING GEAR CROSS TUBE FAIRINGS REMOVED

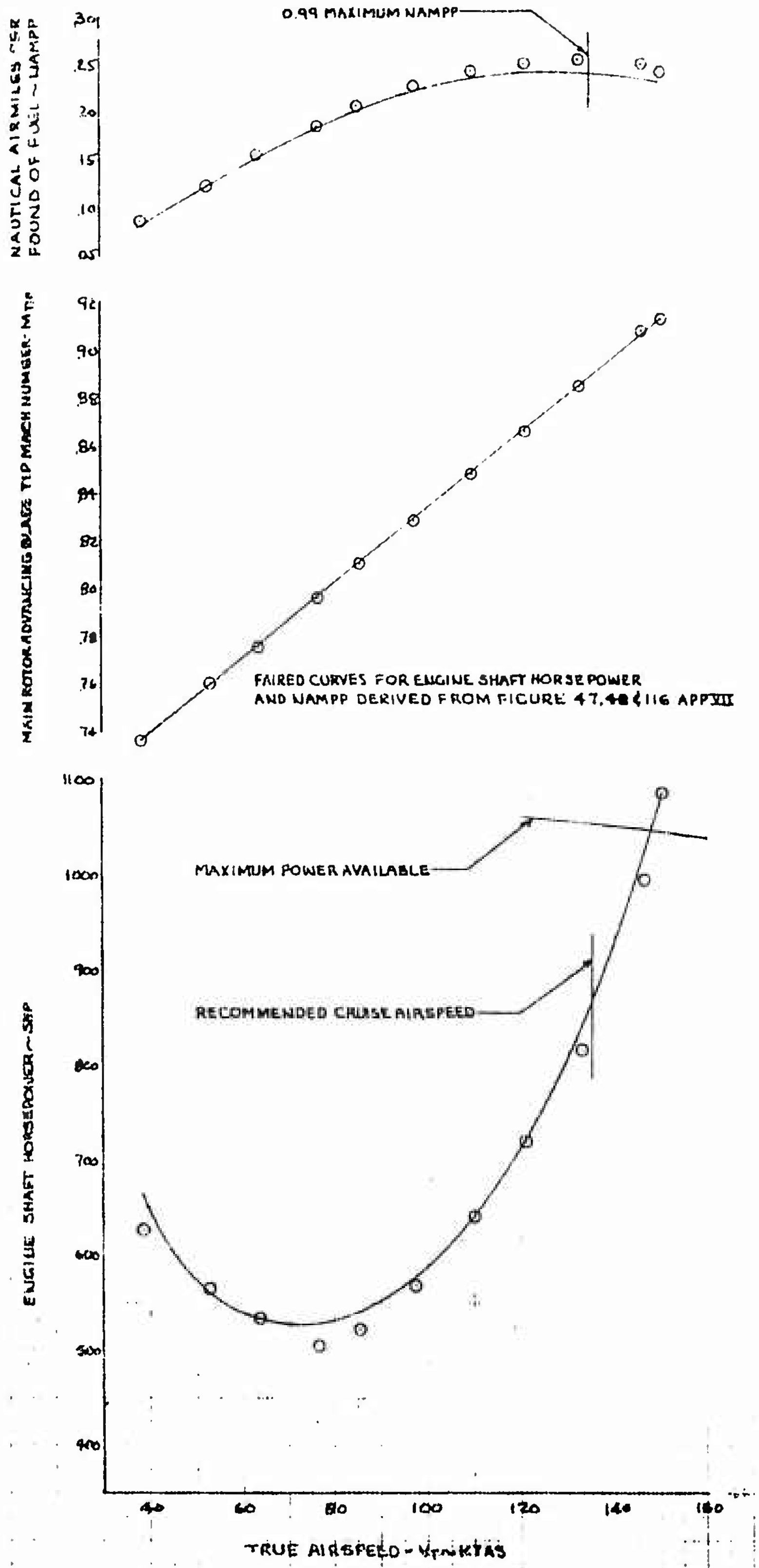


FIGURE NO. 51

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

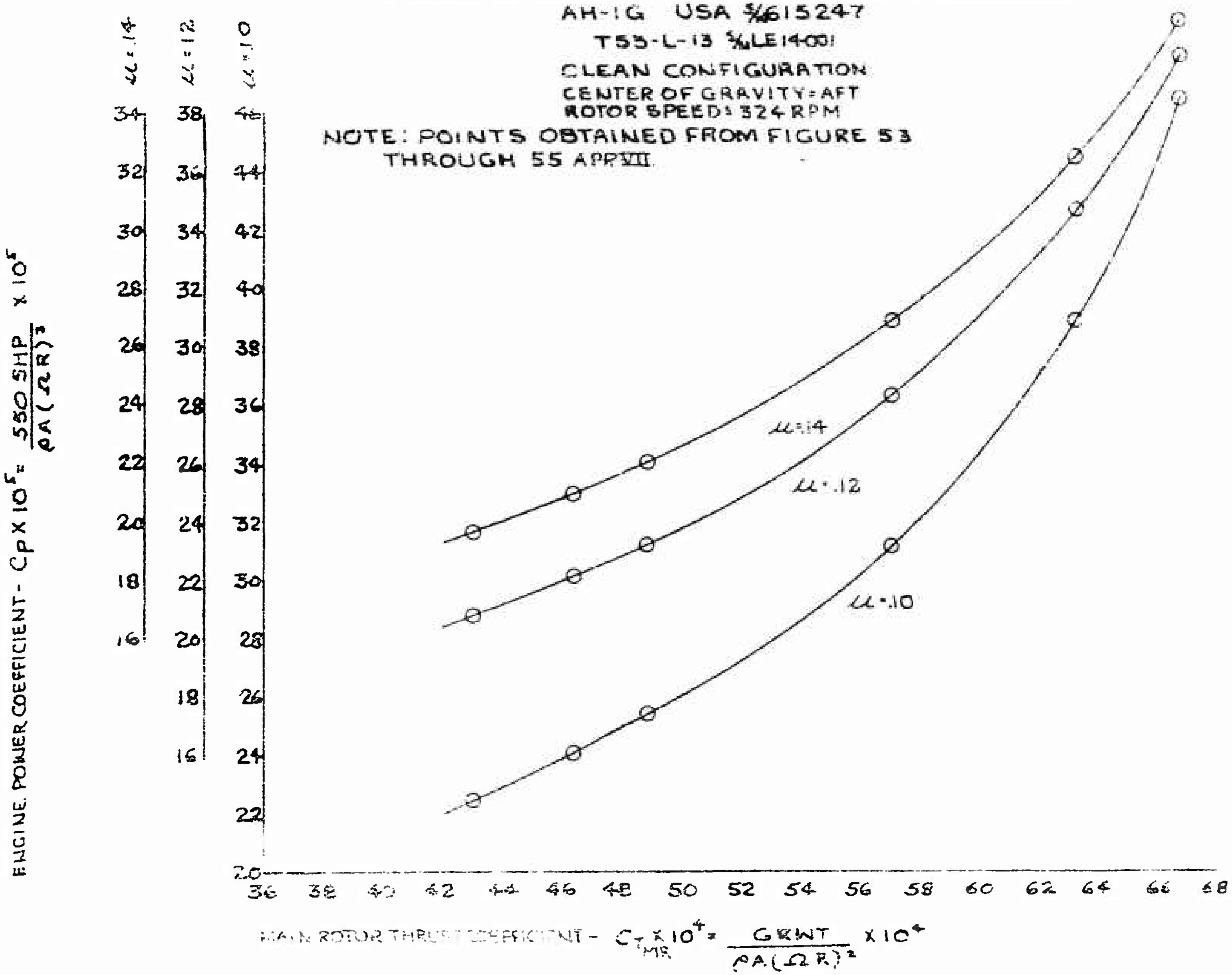


FIGURE No 52

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

CLEAN CONFIGURATION
CENTER OF GRAVITY: AFT
ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 53 THROUGH 55 APP VII

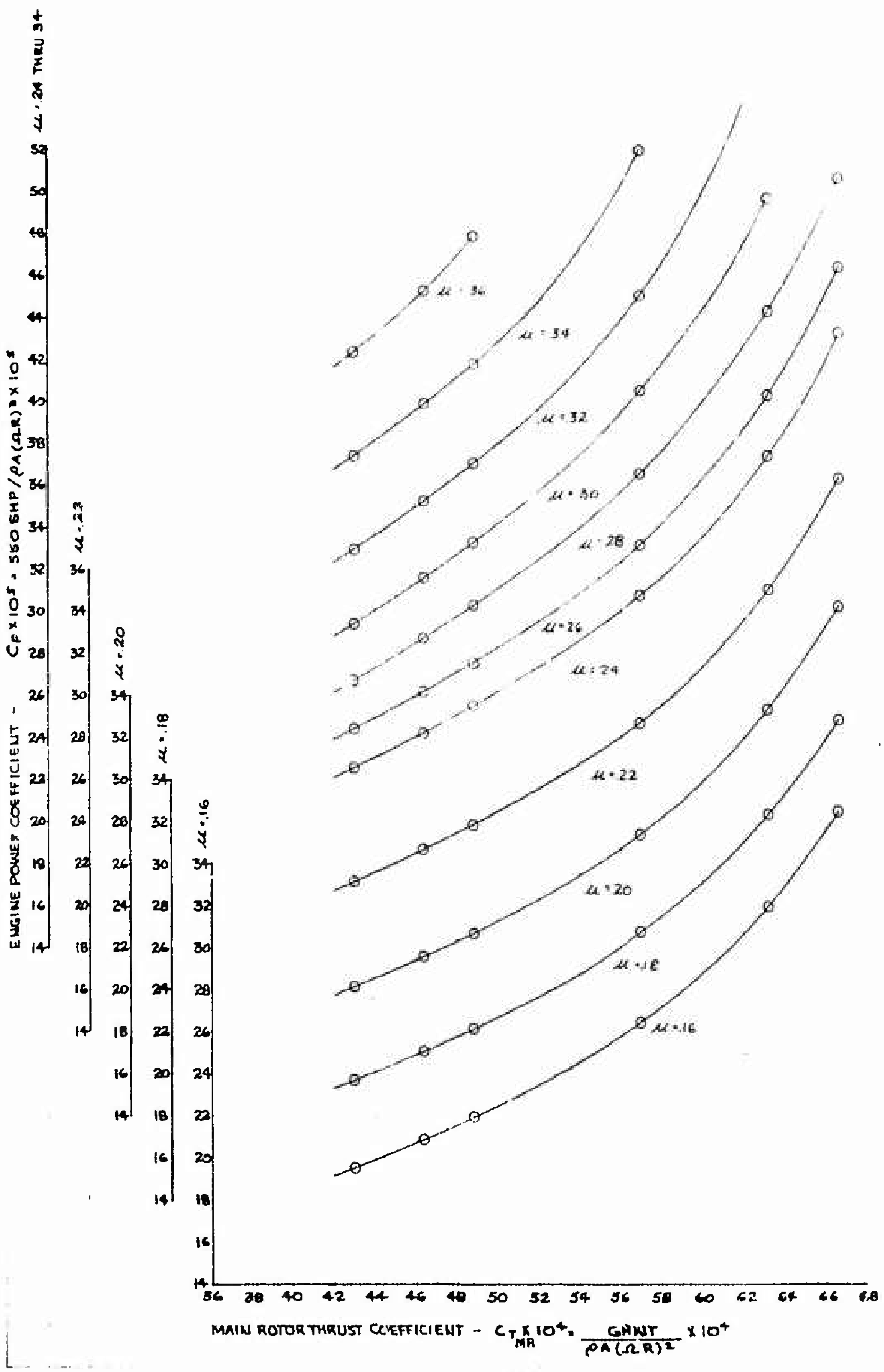


FIGURE NO 53

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB.	AVG. LONG C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	4270	7600	201.1 (AFT)	0.004290	323.5	CLEAN
□	4700	8100	200.6 (AFT)	0.004640	323.5	CLEAN

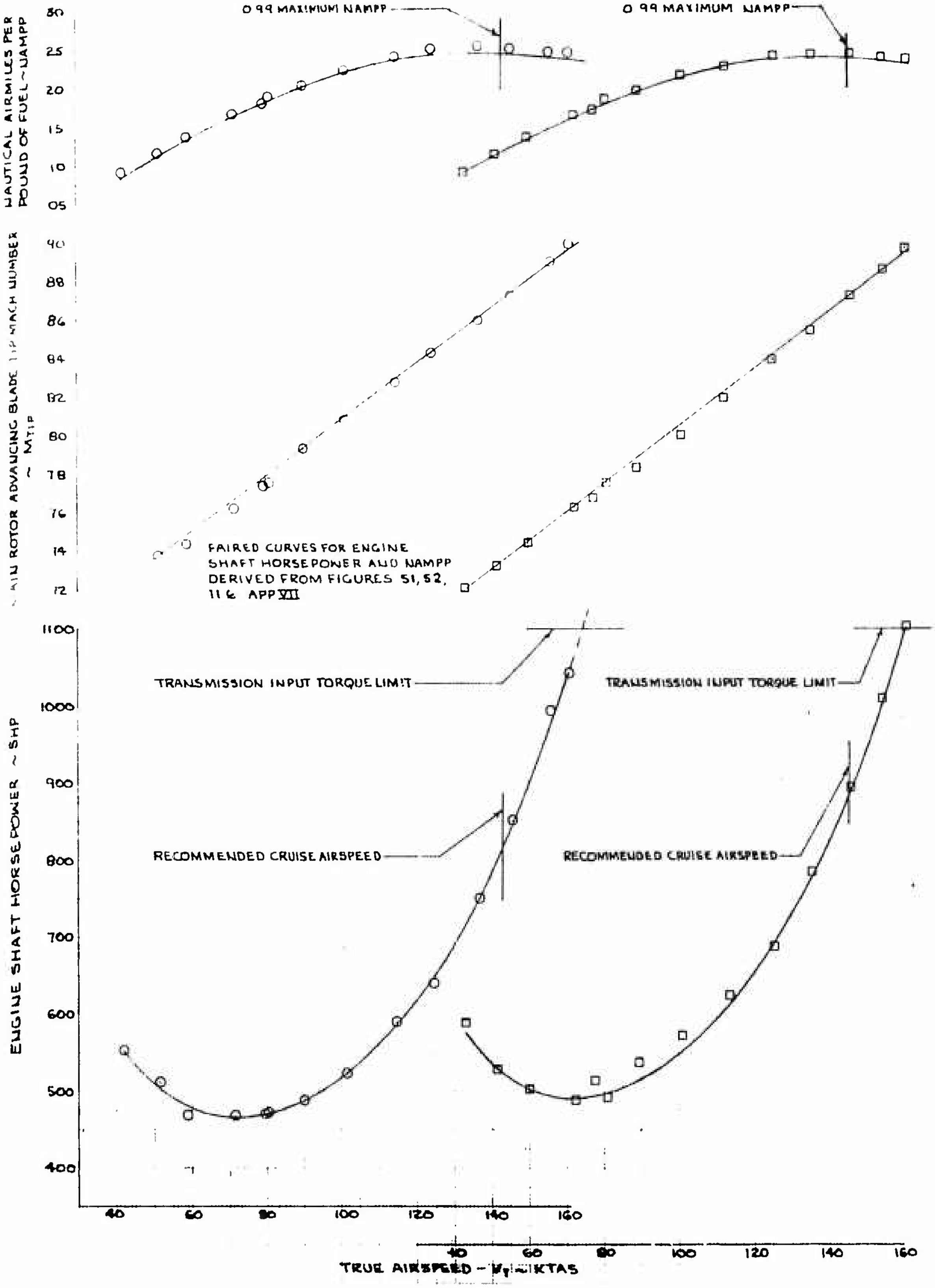


FIGURE NO 54

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 66-15247

T53-L-13 S/N LE14-001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G ~ IN.	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	4700	8470	203.3 (AFT)	0.004882	323.5	CLEAN
□	4550	8570	200.7 (AFT)	0.005699	323.5	CLEAN

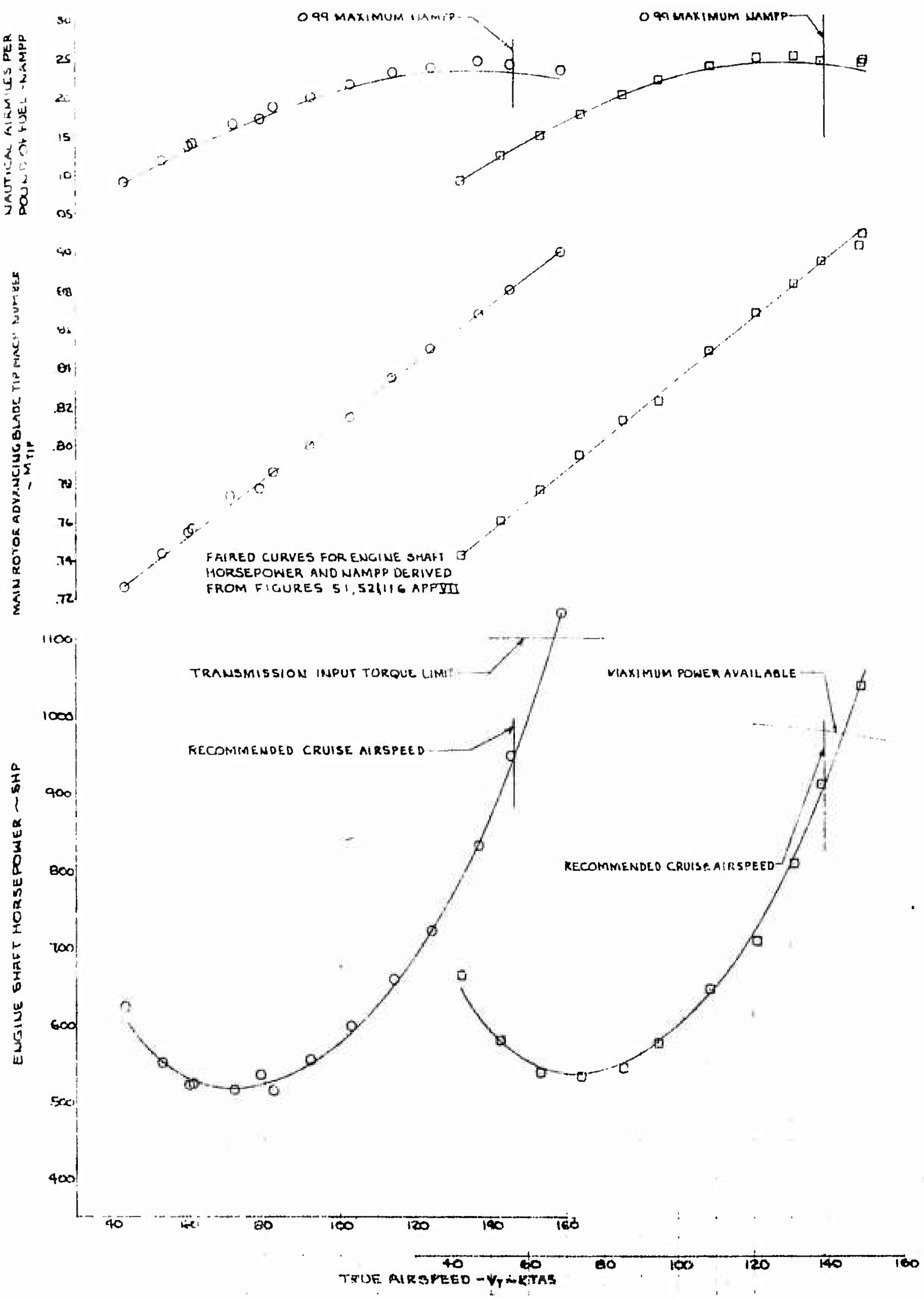


FIGURE NO 55
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C G ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	14410	8140	200 8(AFT)	0 006321	323 5	CLEAN
□	14790	8480	200.3(AFT)	0.006667	323 5	CLEAN

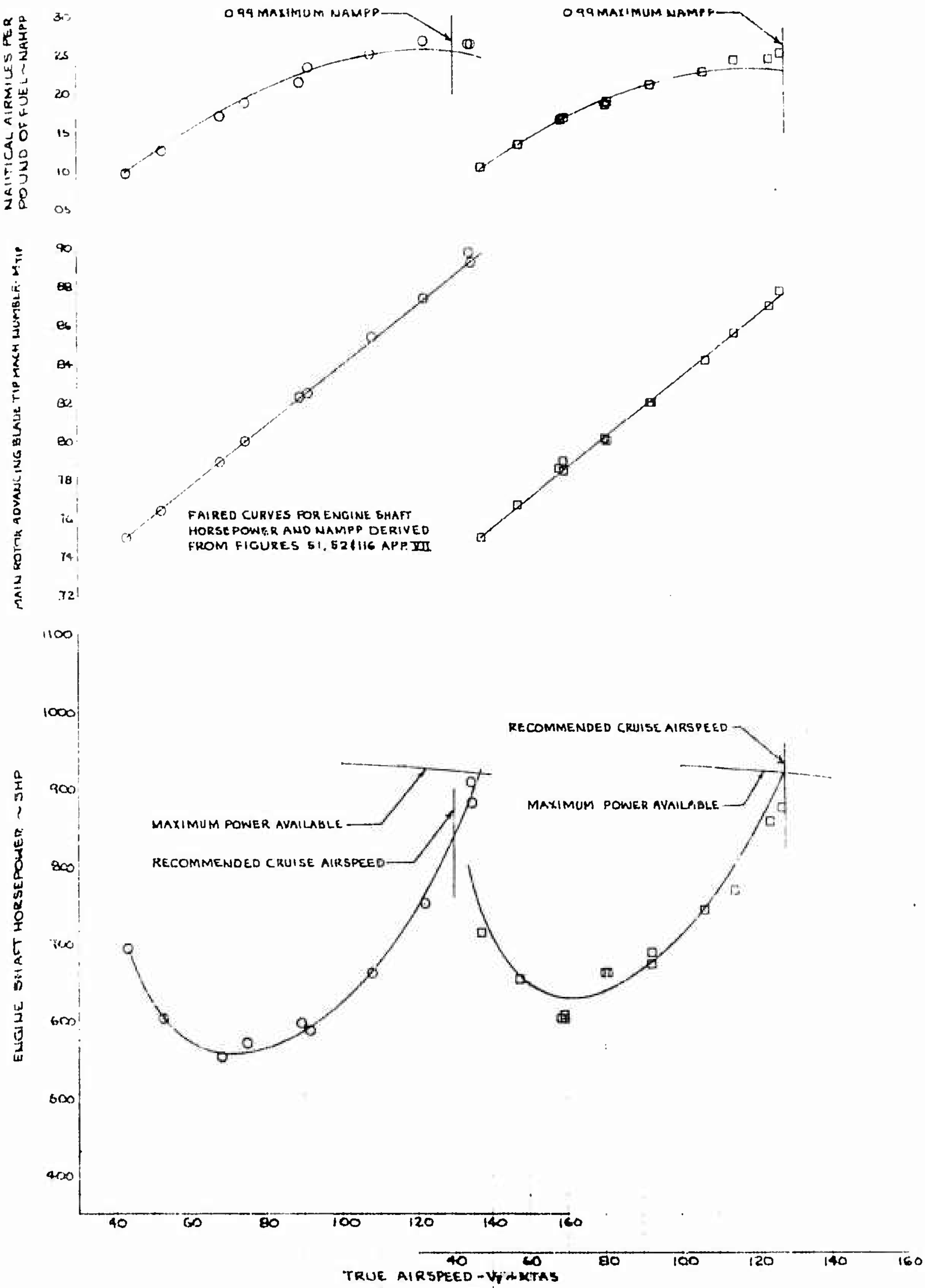


Figure No. 56

Non Dimensional Level Flight Performance

AH-1G USA S/N 615247
T53-L-13 S/N LE 14001
BASIC CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED
CENTER OF GRAVITY - FORWARD
ROTOR SPEED: 324 RPM
NOTE: POINTS OBTAINED FROM FIGURES 58 AND 59 APP VII

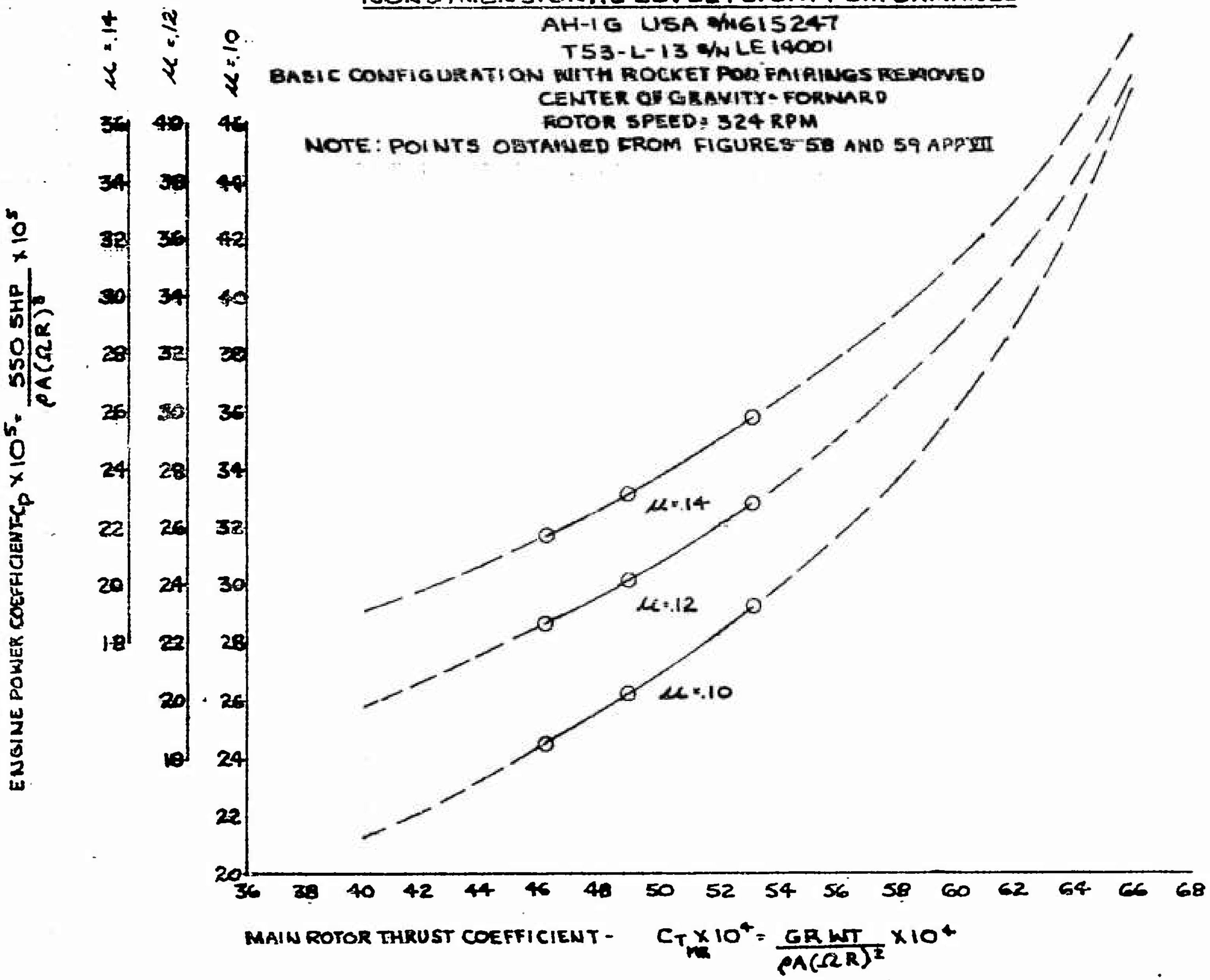


FIGURE NO 57

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA 66-15247

T53-L-13 S/N LE 14001

BASIC CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY = FORWARD

ROTOR SPEED = 324 RPM

NOTE POINTS OBTAINED FROM FIGURE 58 AND 59 APP II

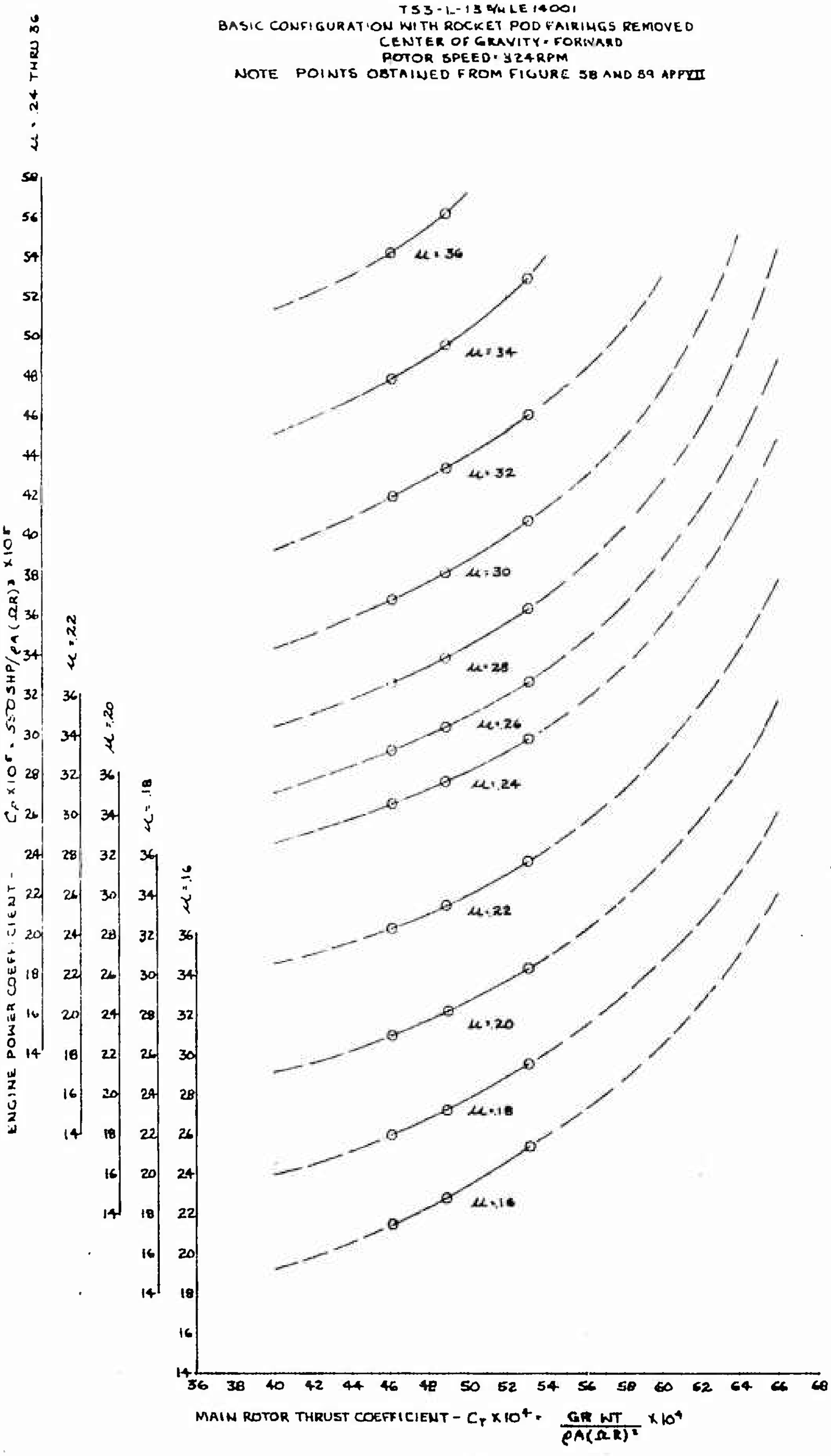


FIGURE NO 58

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	4220	8170	141.9 (FWD)	0.004613	323.5	BASIC
□	4220	8660	192.5 (FWD)	0.004889	323.5	BASIC

NOTE: ALL ROCKET POD FAIRINGS REMOVED

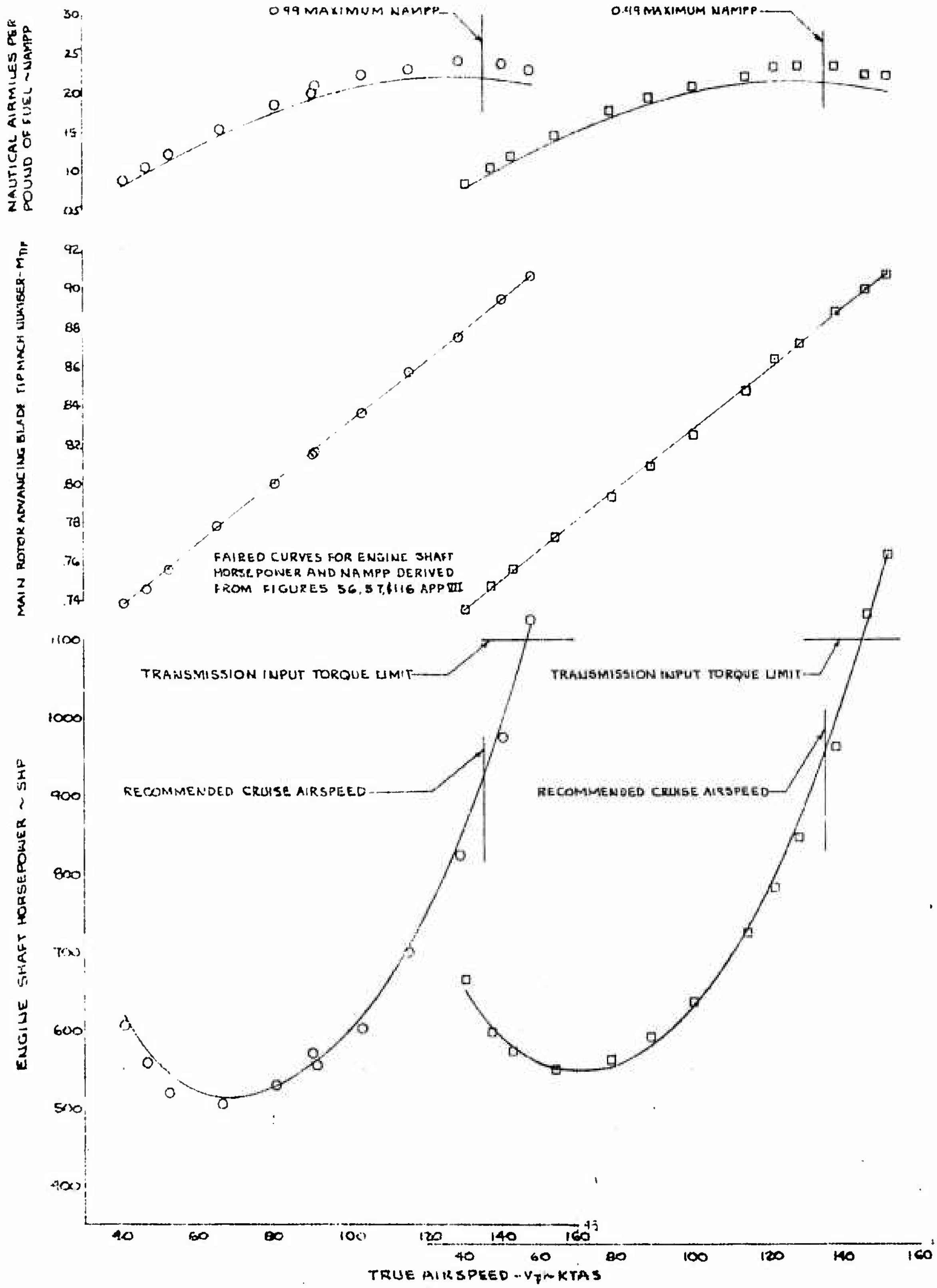


FIGURE NO 59

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14-001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G ~ IN	AVG THRUST COEFF. ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	9760	7970	192.4 (FWD)	0.005319	324	BASIC

NOTE: ALL ROCKET POD FAIRINGS REMOVED

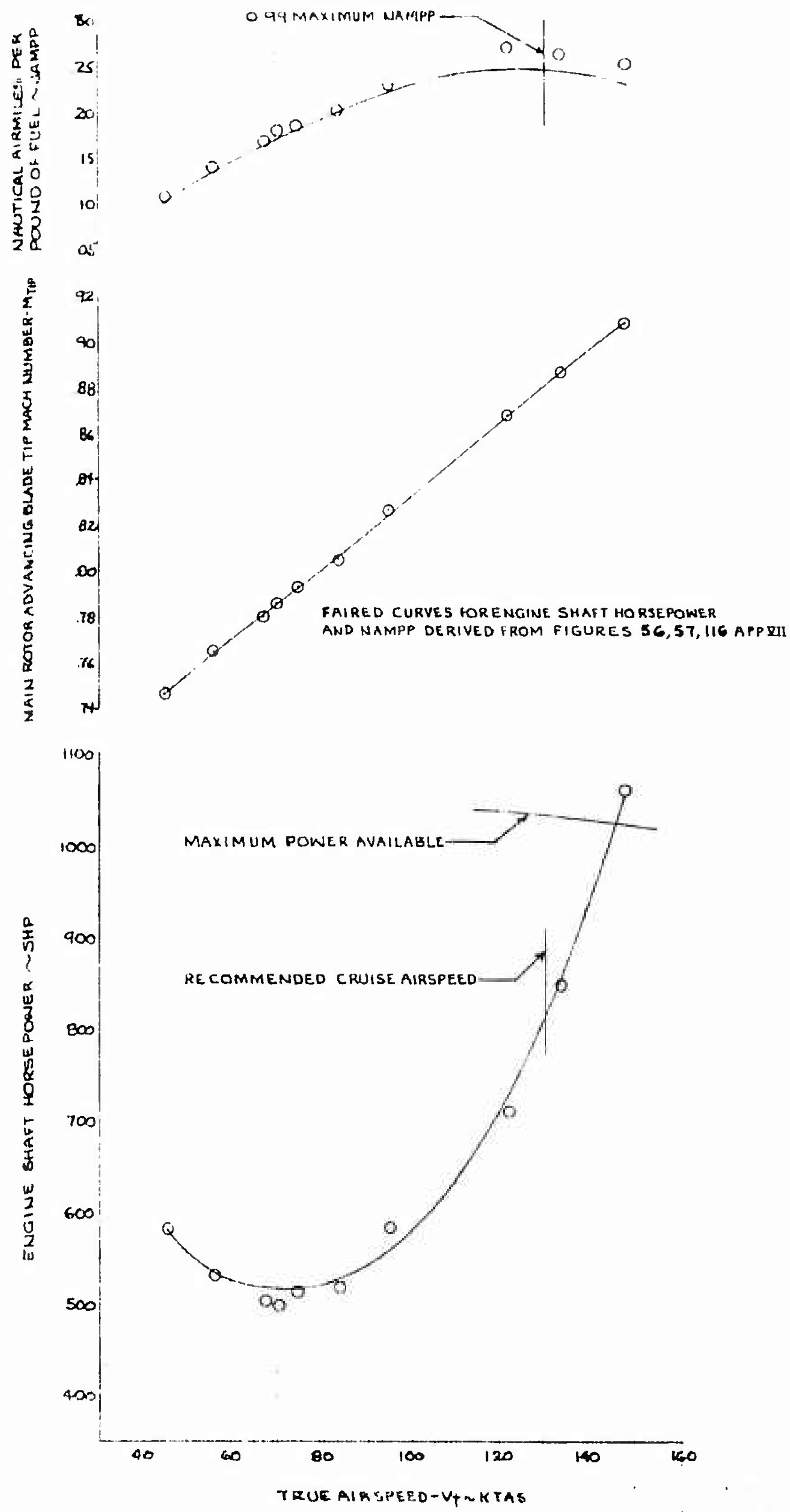


FIGURE NO. 60

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

BASIC CONFIGURATION WITH ROCKET POD FAIRINGS INSTALLED

CENTER OF GRAVITY = FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 62 AND 63 APP VII

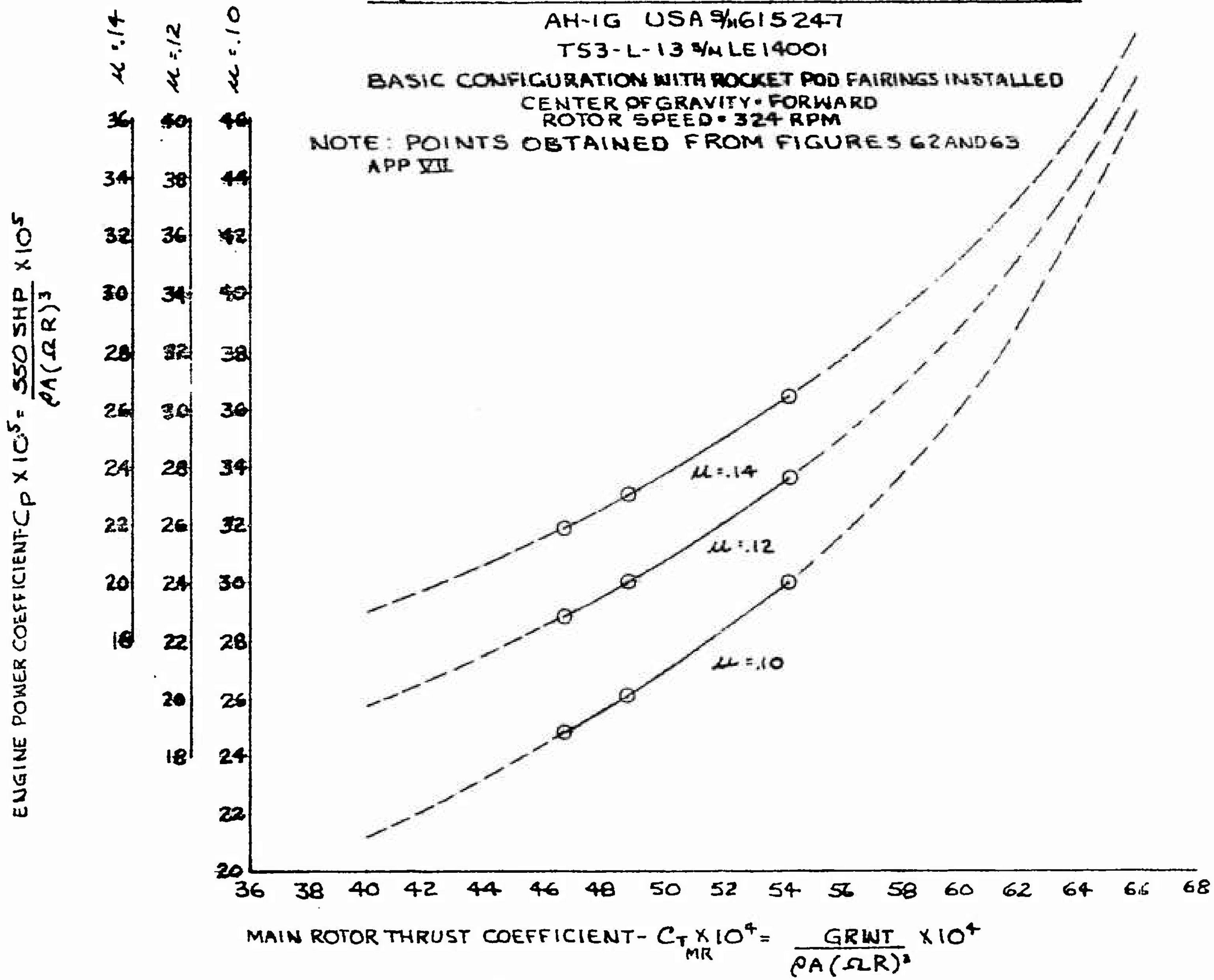


FIGURE NO 61

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE 14001

BASIC CONFIGURATION WITH ROCKET POD FAIRINGS INSTALLED

CENTER OF GRAVITY - FORWARD

ROTOR SPEED - 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 62 AND 63 APP VII

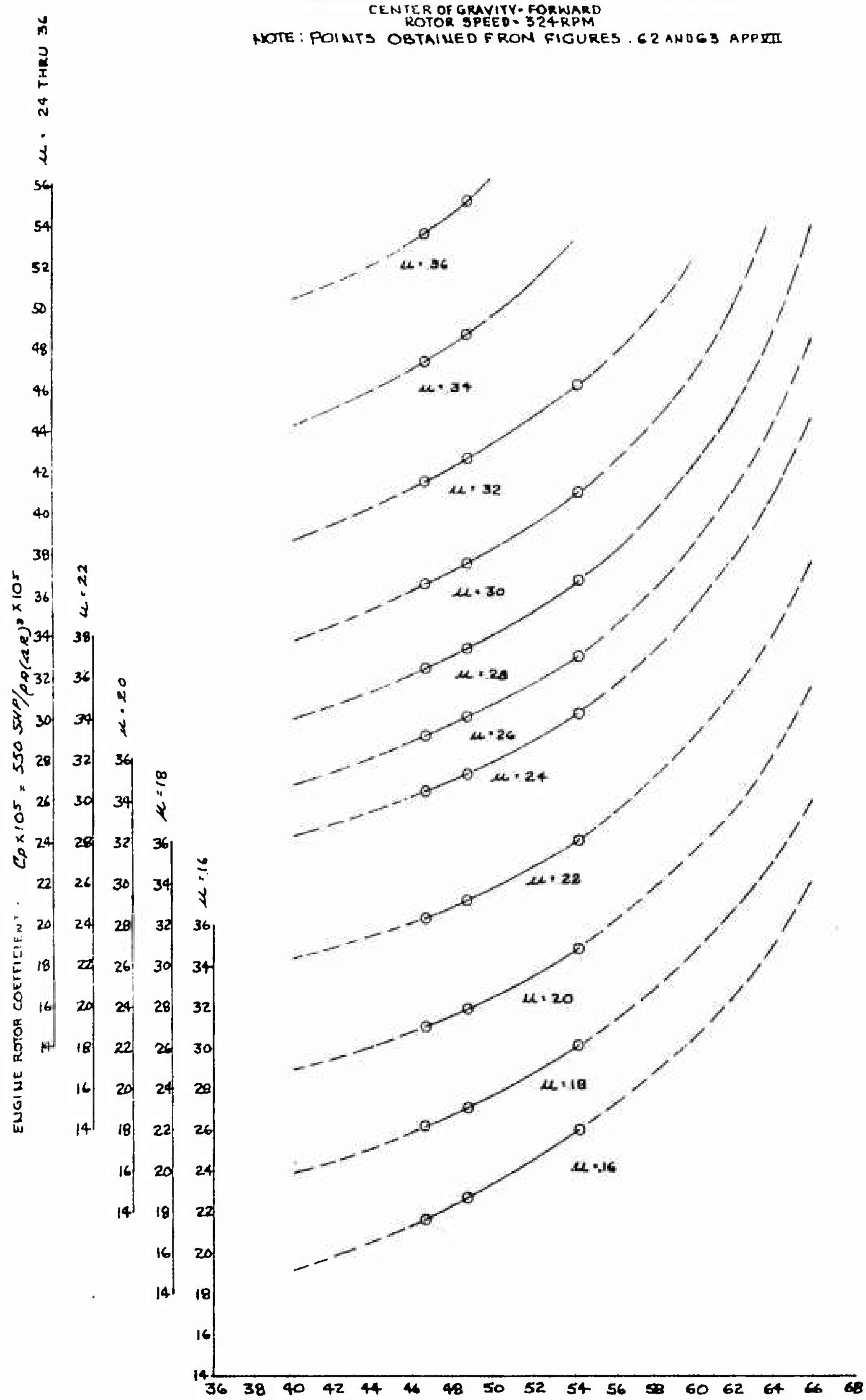


FIGURE NO. 62

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE 14001

SYMBOL	AVG. ALTITUDE H_D ~ FEET	AVG. GROSS WEIGHT ~ LB	AVG. LONG. CG. ~ IN.	AVG. THRUST COEFF. ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	3880	8680	191.3(FWD)	0.004880	322.5	BASIC
□	3850	8270	191.0(FWD)	0.004661	322.0	BASIC

NOTE: ALL ROCKET POD FAIRINGS INSTALLED

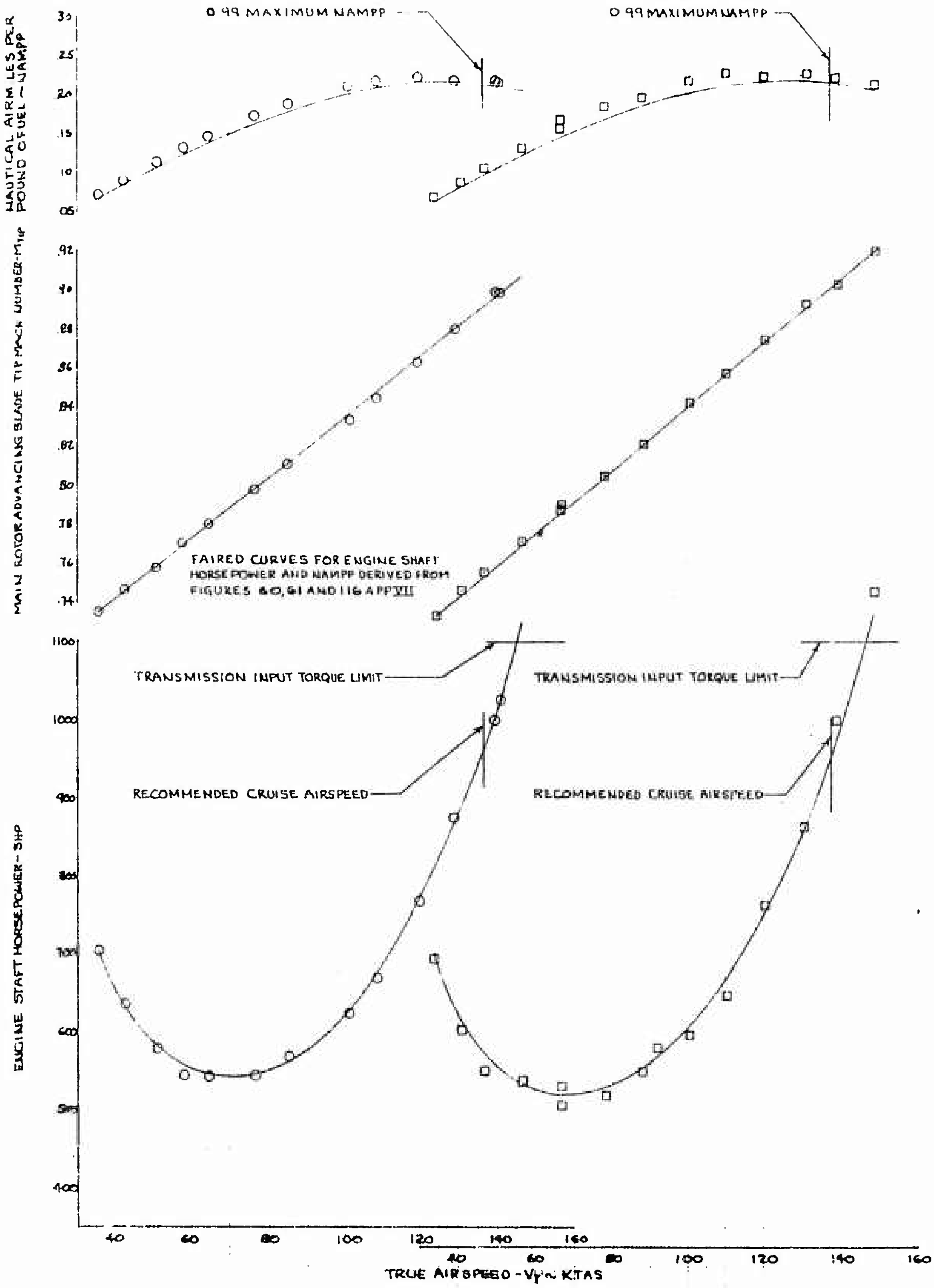


FIGURE NO 63

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	AVG ALTITUDE H_D ~ FEET	AVG GROSS WEIGHT ~ LB	AVG. LONG C.G. ~ IN.	AVG THRUST COEFF. ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
O	10540	7850	190.6 (FWD)	0.005419	322.5	BASIC

NOTE: ALL ROCKET POD FAIRINGS INSTALLED

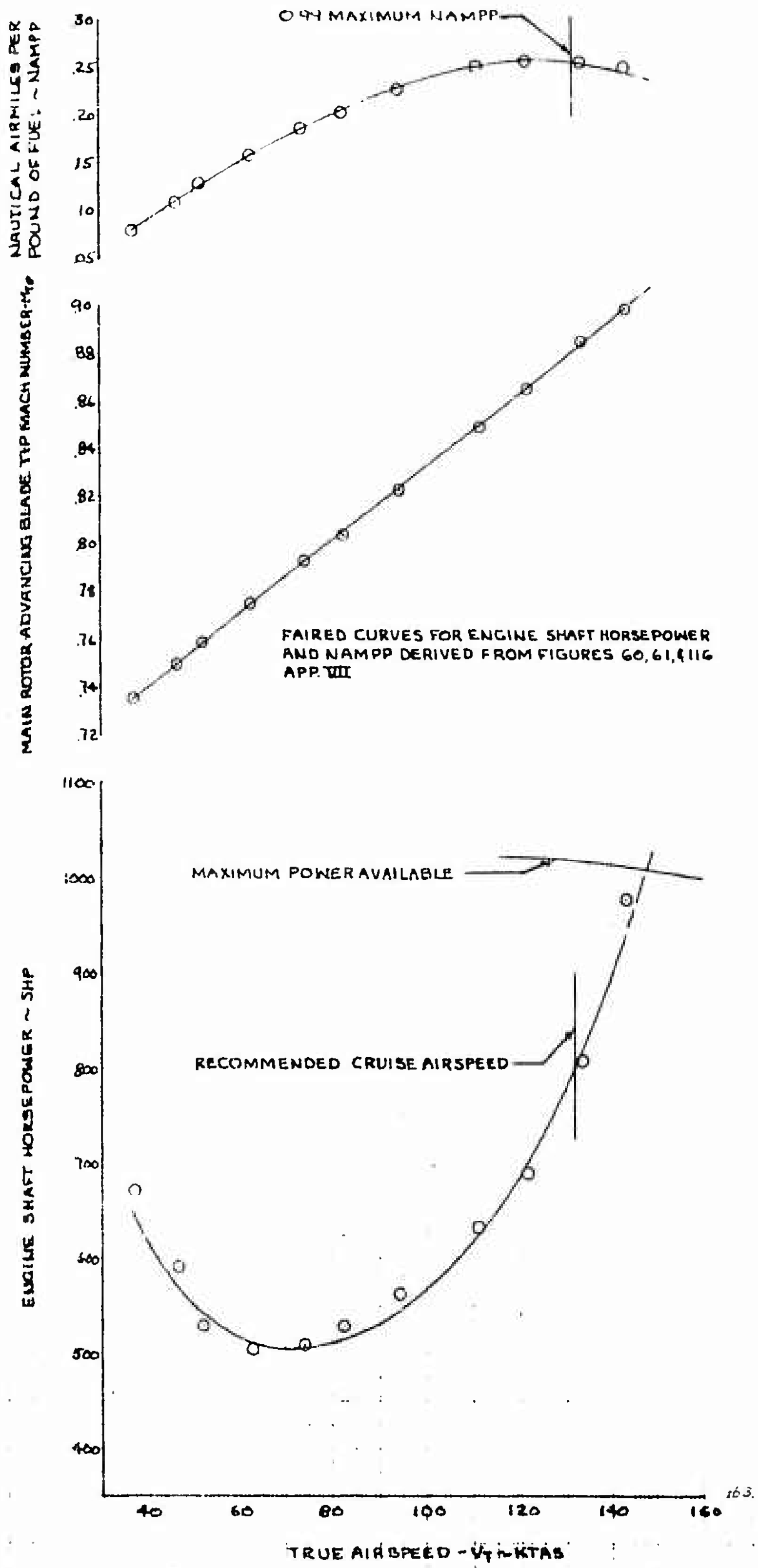


FIGURE No. 64

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA SN 615247

T53-L-13 SN LE 14001

LT. SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY: FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 66 AND 67 APP XII

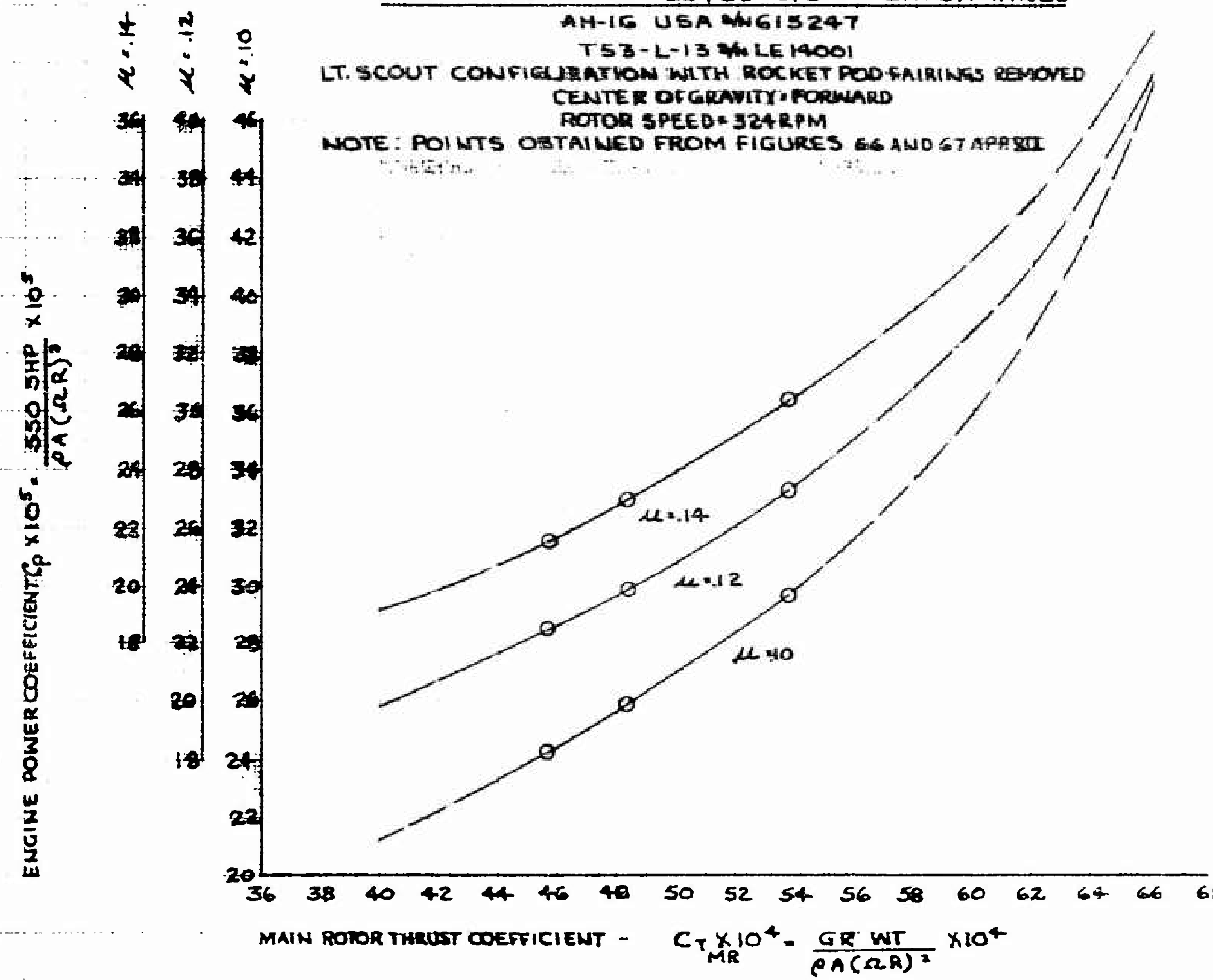


FIGURE NO. 65

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE 14001

LT. SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY - FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 66 AND 67 APP III

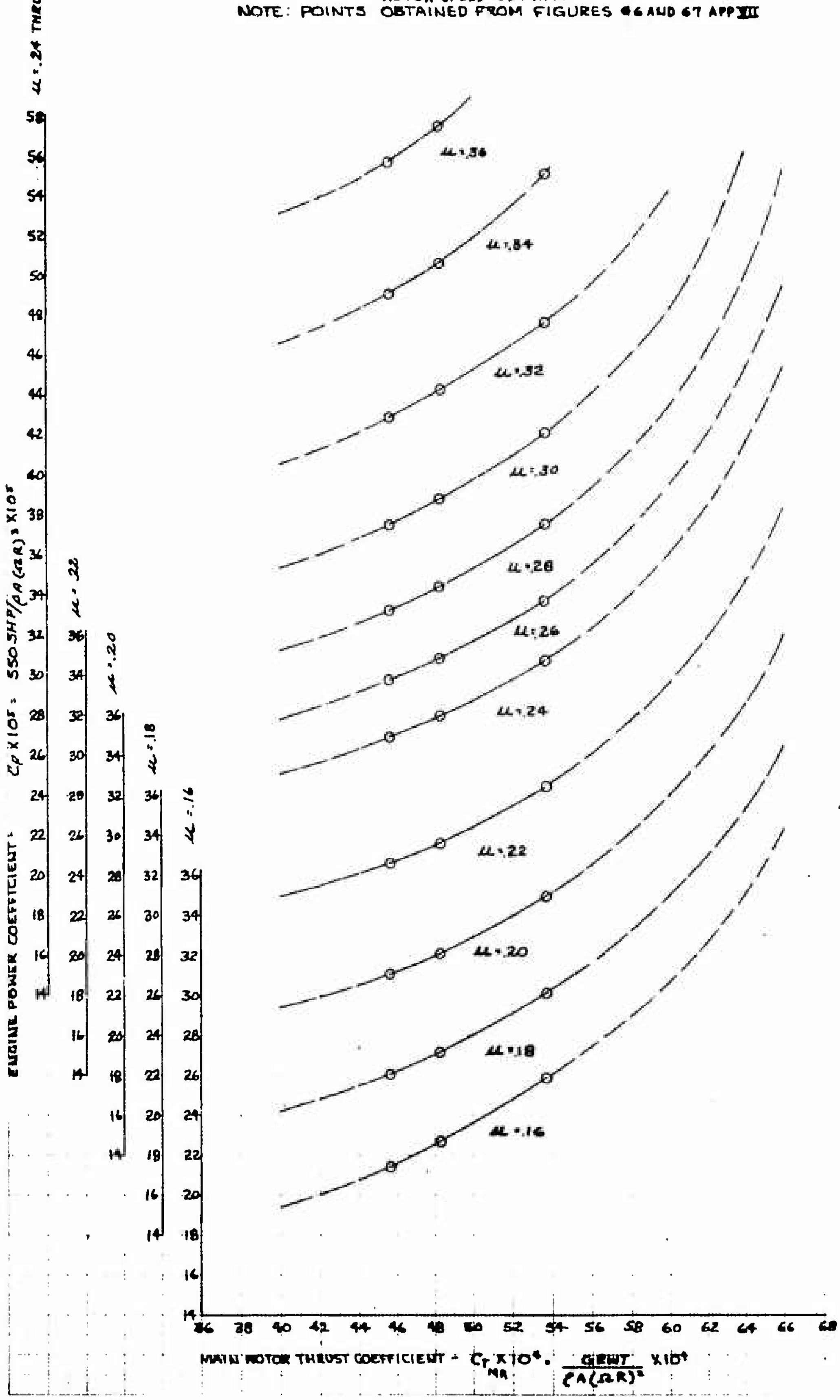


FIGURE NO 66
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	4160	8170	193.2(FWD)	0.004562	325	LT SCOUT
□	4300	8640	192.2(FWD)	0.004830	325.5	LT SCOUT

NOTE: ALL ROCKET POD FAIRINGS REMOVED

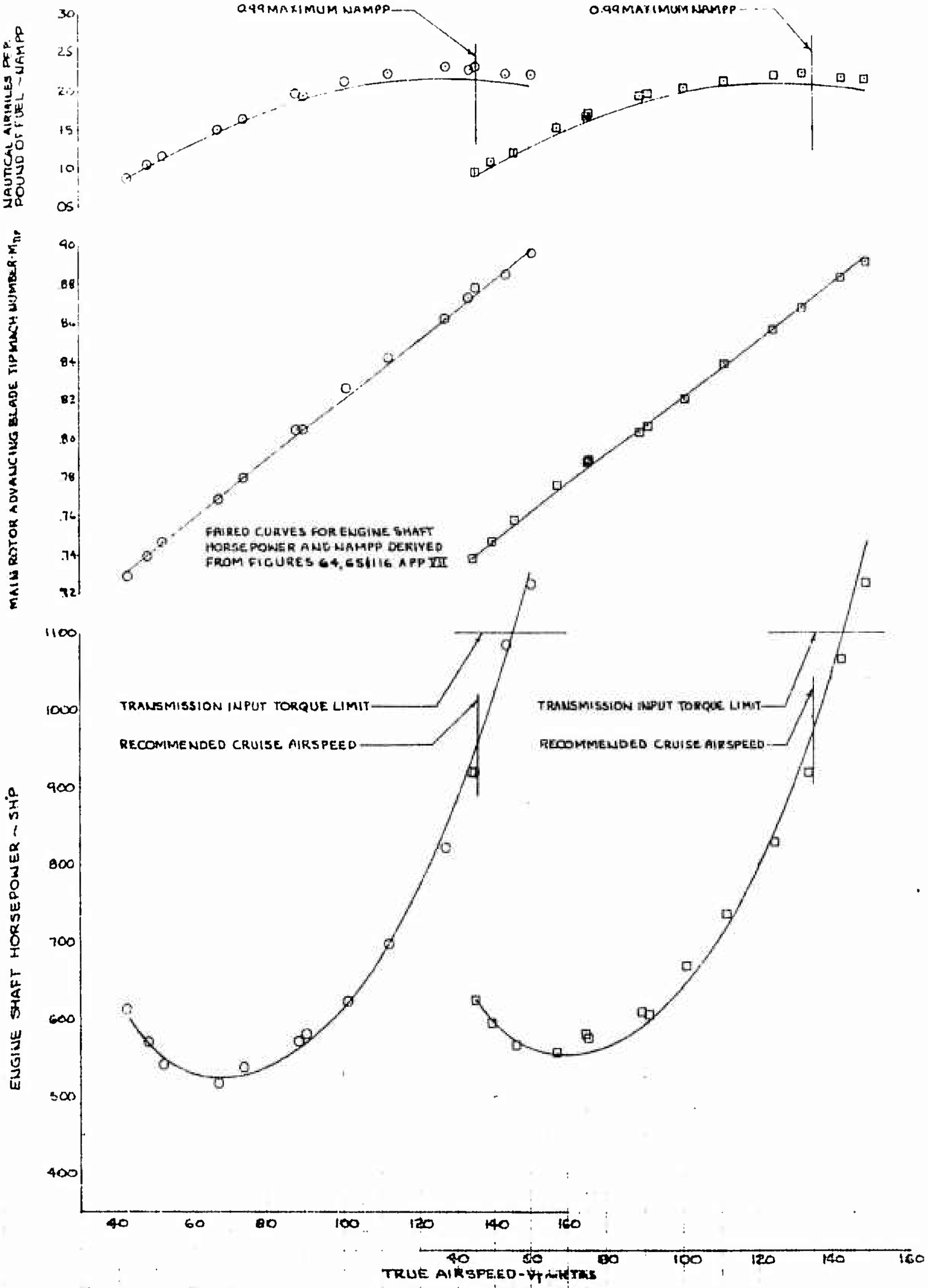


FIGURE NO 67
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
O	9260	8150	194.5 (FWD)	0.005371	323.5	LT SCOUT

NOTE: ALL ROCKET POD FAIRINGS REMOVED

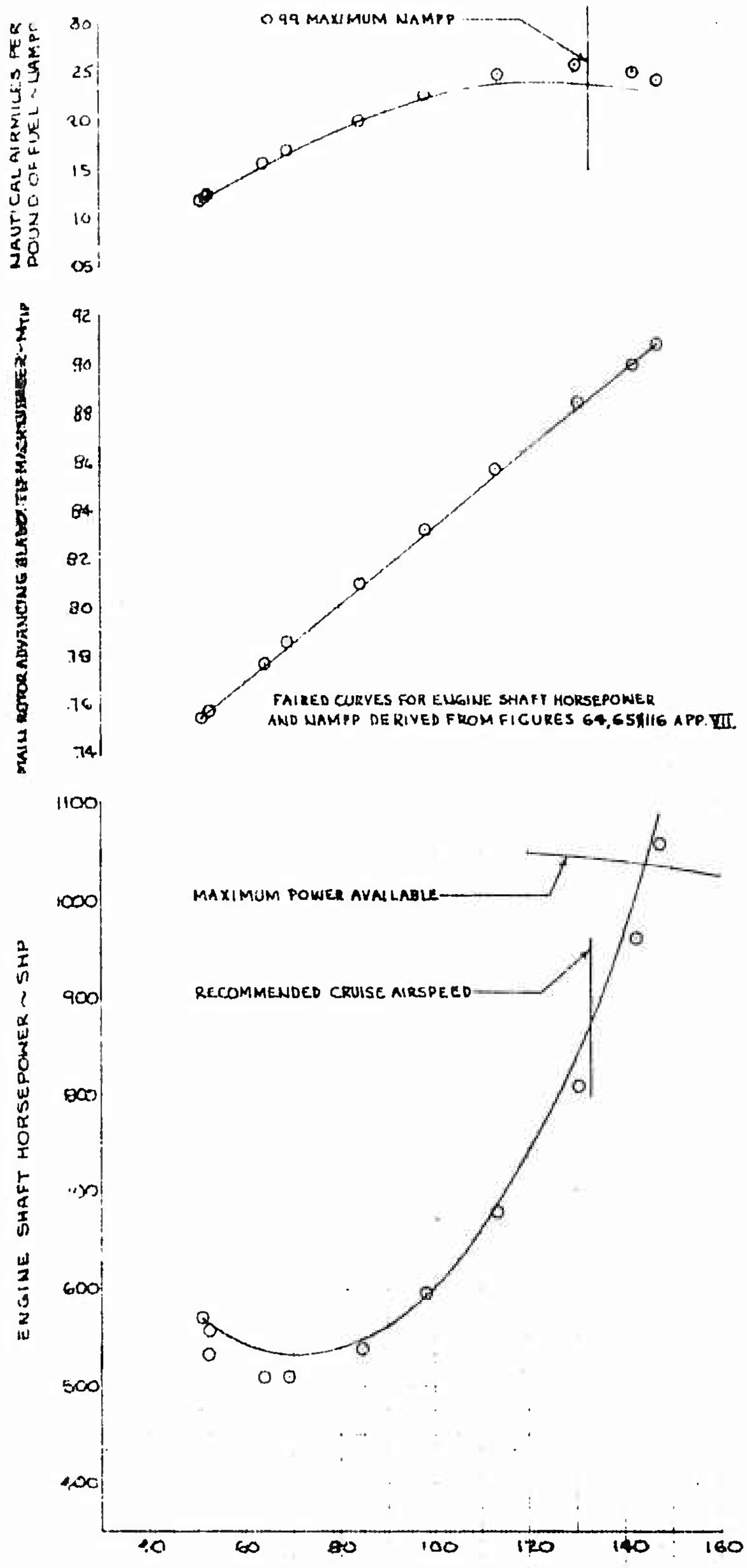


FIGURE NO. 68

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

MIB'D ALTERNATE CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY - FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 70 AND 71 APP VII

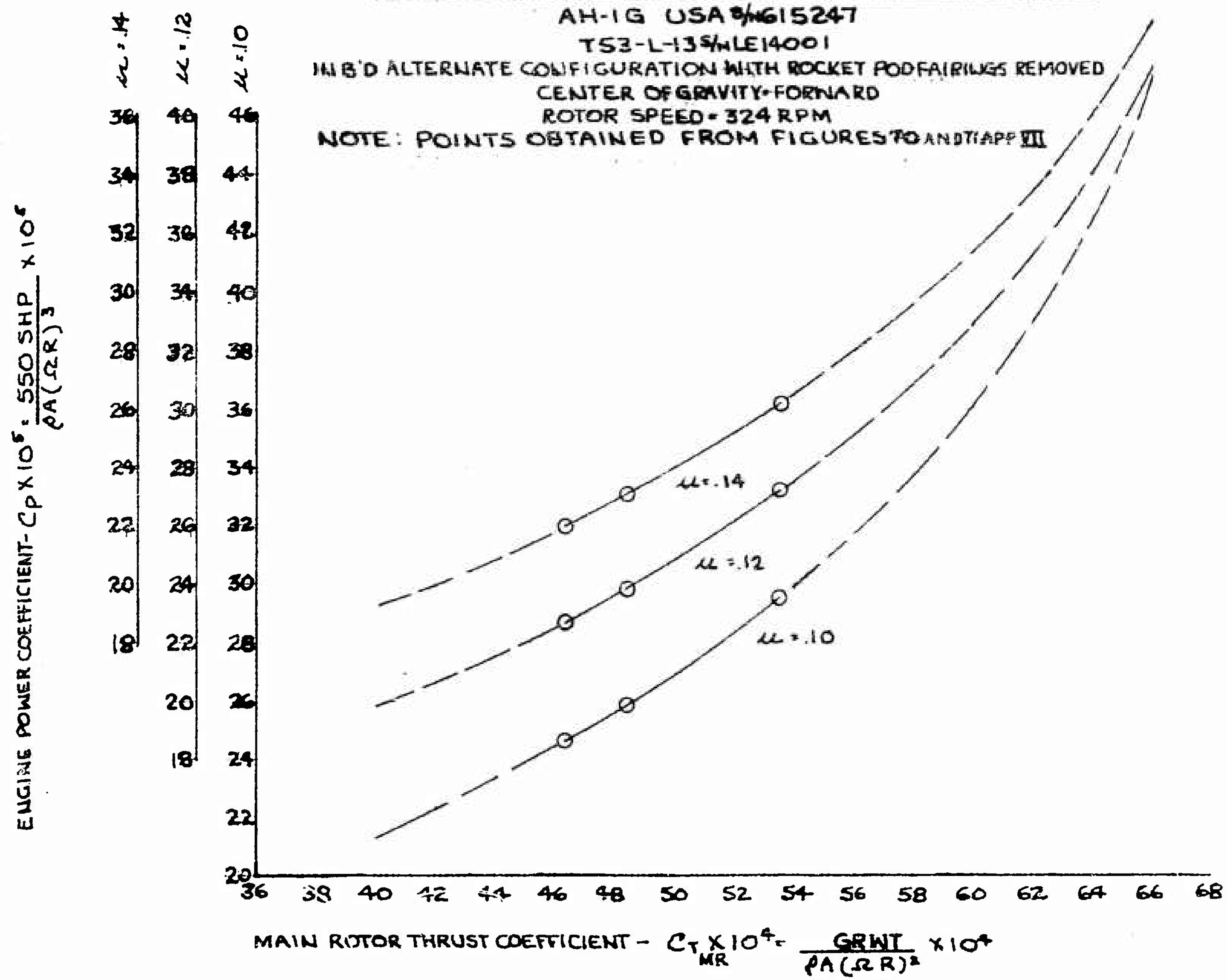


FIGURE NO 69

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

MI B'D ALTERNATE CONFIGURATION WITH ROCKET PODFAIRINGS REMOVED

CENTER OF GRAVITY: FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 70 AND 71 APP III

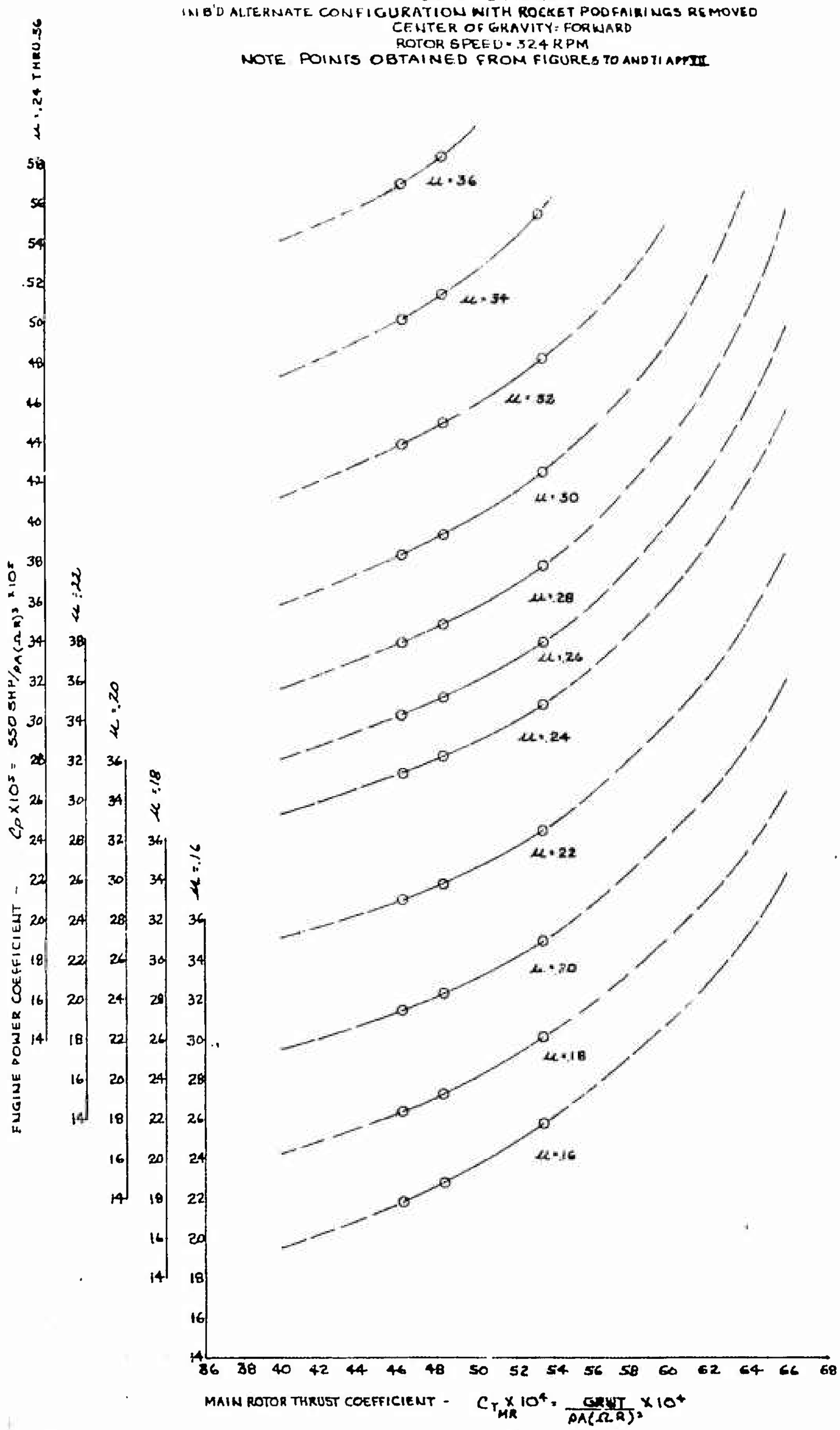


FIGURE No. 70

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG. ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	3960	8290	192.1 (FWD)	0.004630	324	INBD ALTERNATE
□	[illegible]	8770	192.8 (FWD)	0.004838	323.5	INBD ALTERNATE

NOTE ALL ROCKET POD FAIRINGS REMOVED

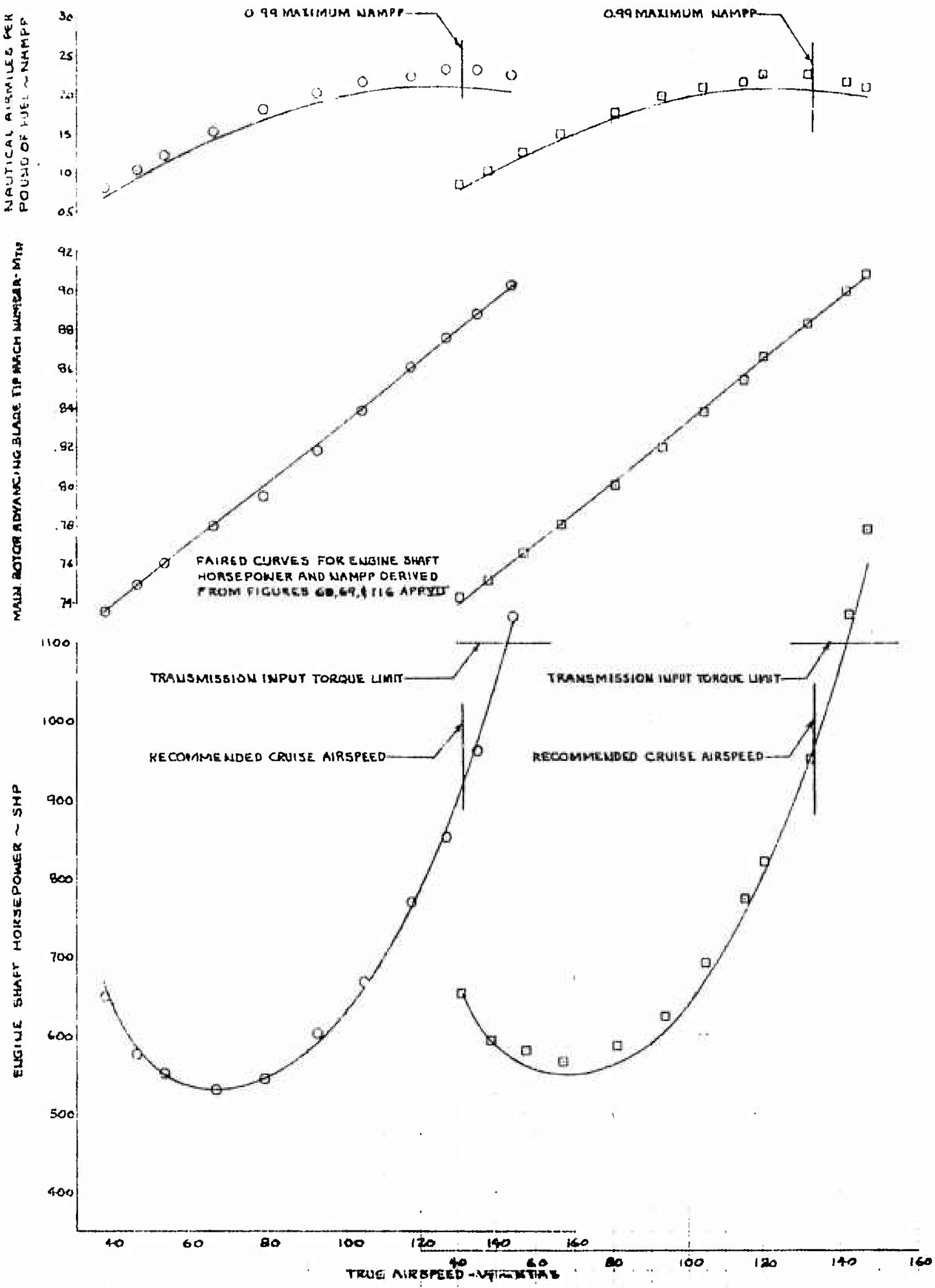


FIGURE NO 71
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_p ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG. C.G. ~ IN	AVG. THRUST COEFF.	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	9630	8050	193.8 (FWD)	0.005351	324	INB'D ALTERNATE

NOTE: ALL ROCKET POD FAIRINGS REMOVED

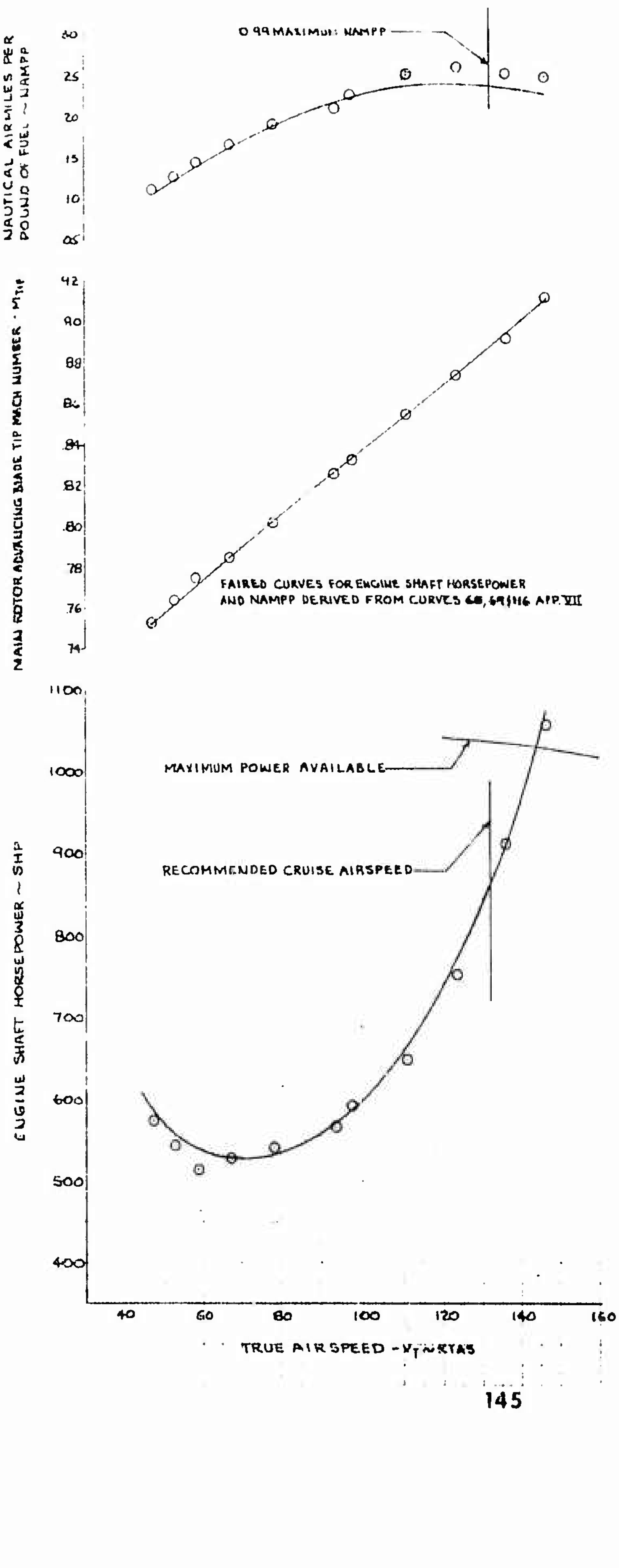


FIGURE NO. 72

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

OUTBD ALTERNATE CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY - FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 74 AND 75 APP VII

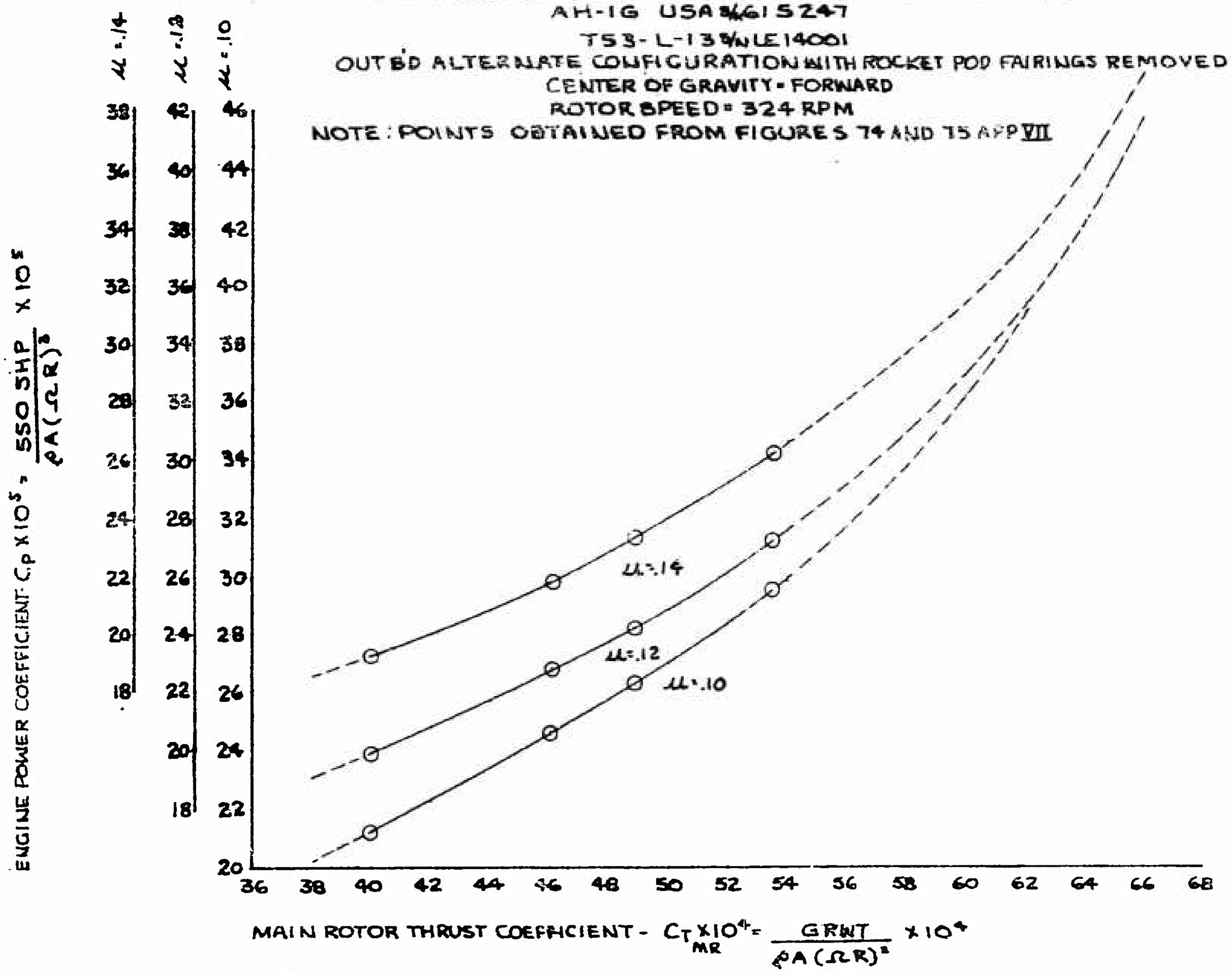


FIGURE NO. 73

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

OUTBD ALTERNATE CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY - FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 74 AND 75 APP VIII

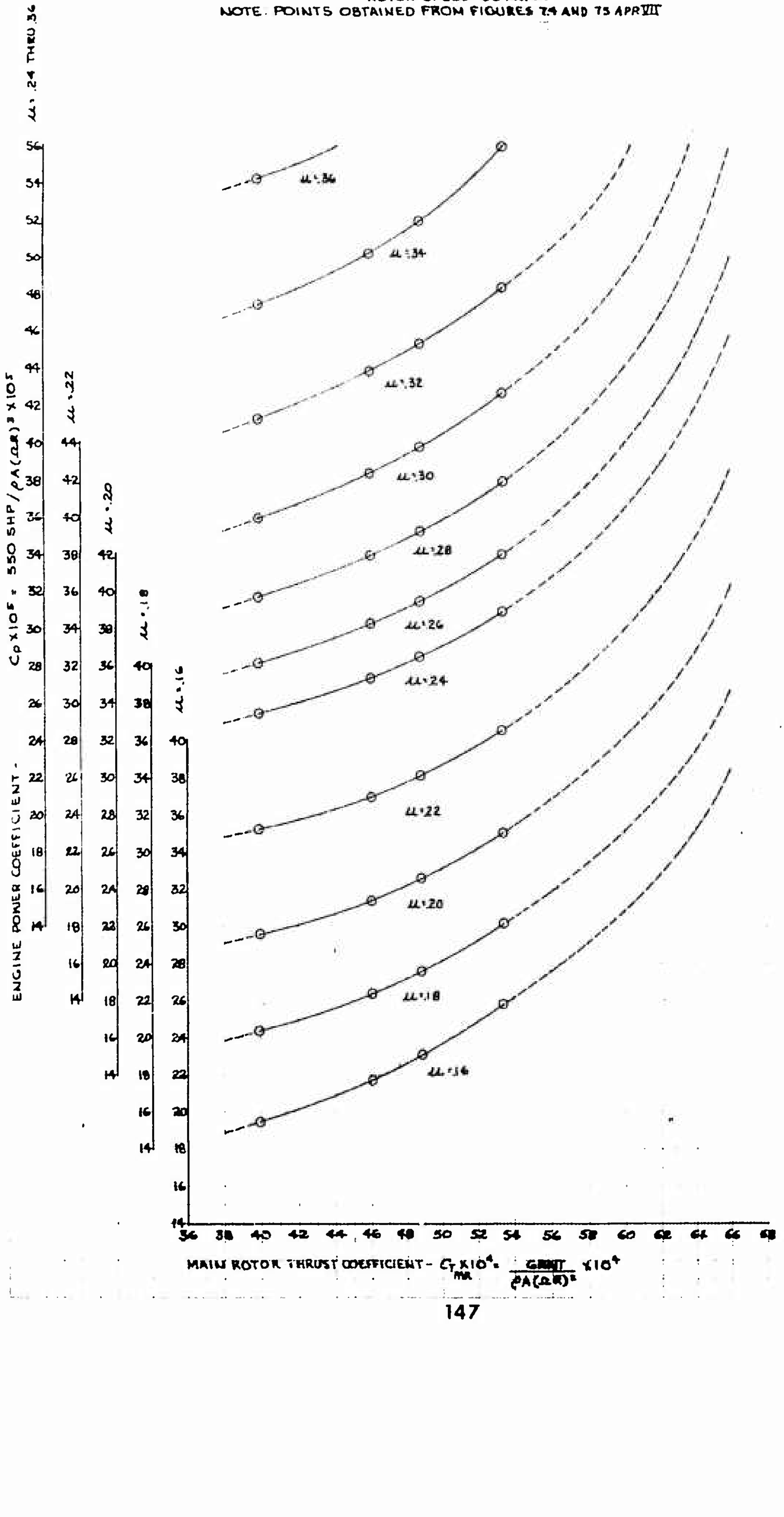


FIGURE No. 74
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 NWLE 14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G. ~ IN	AVG THRUST COEFF. ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	1810	7590	191.9 (FWD)	0.003988	323.5	OUTBD ALTERNATE
□	4090	8190	191.3 (FWD)	0.004606	323.5	OUTBD ALTERNATE

NOTE: ALL ROCKET POD FAIRINGS REMOVED

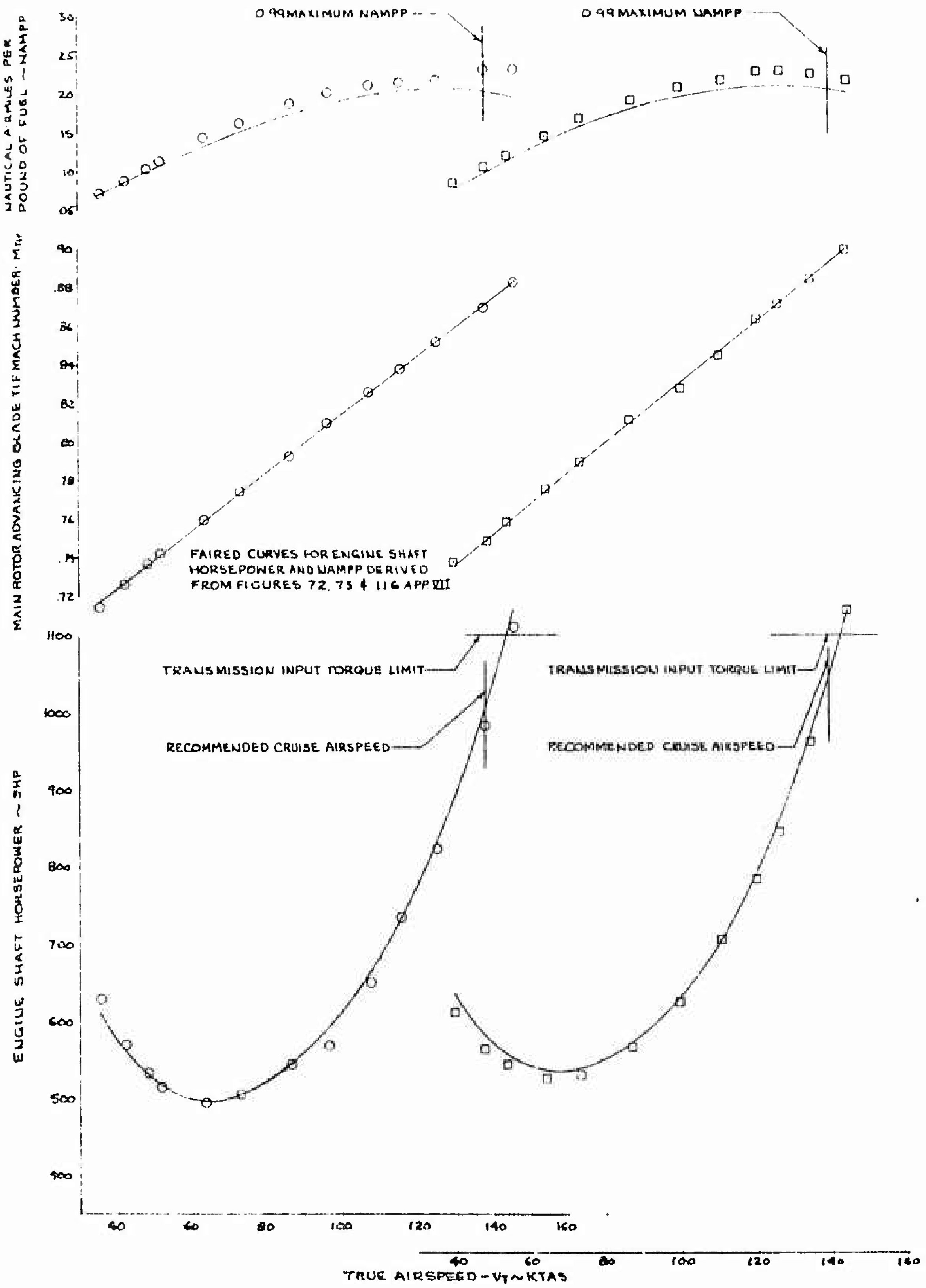


FIGURE NO 75

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG. LONG C.G ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	5960	8720	193.0(FWD)	0.004885	323.5	OUTB'D ALTERNATE
□	9860	8110	191.8(FWD)	0.005346	324	OUTB'D ALTERNATE

NOTE ALL ROCKET POD FAIRINGS REMOVED

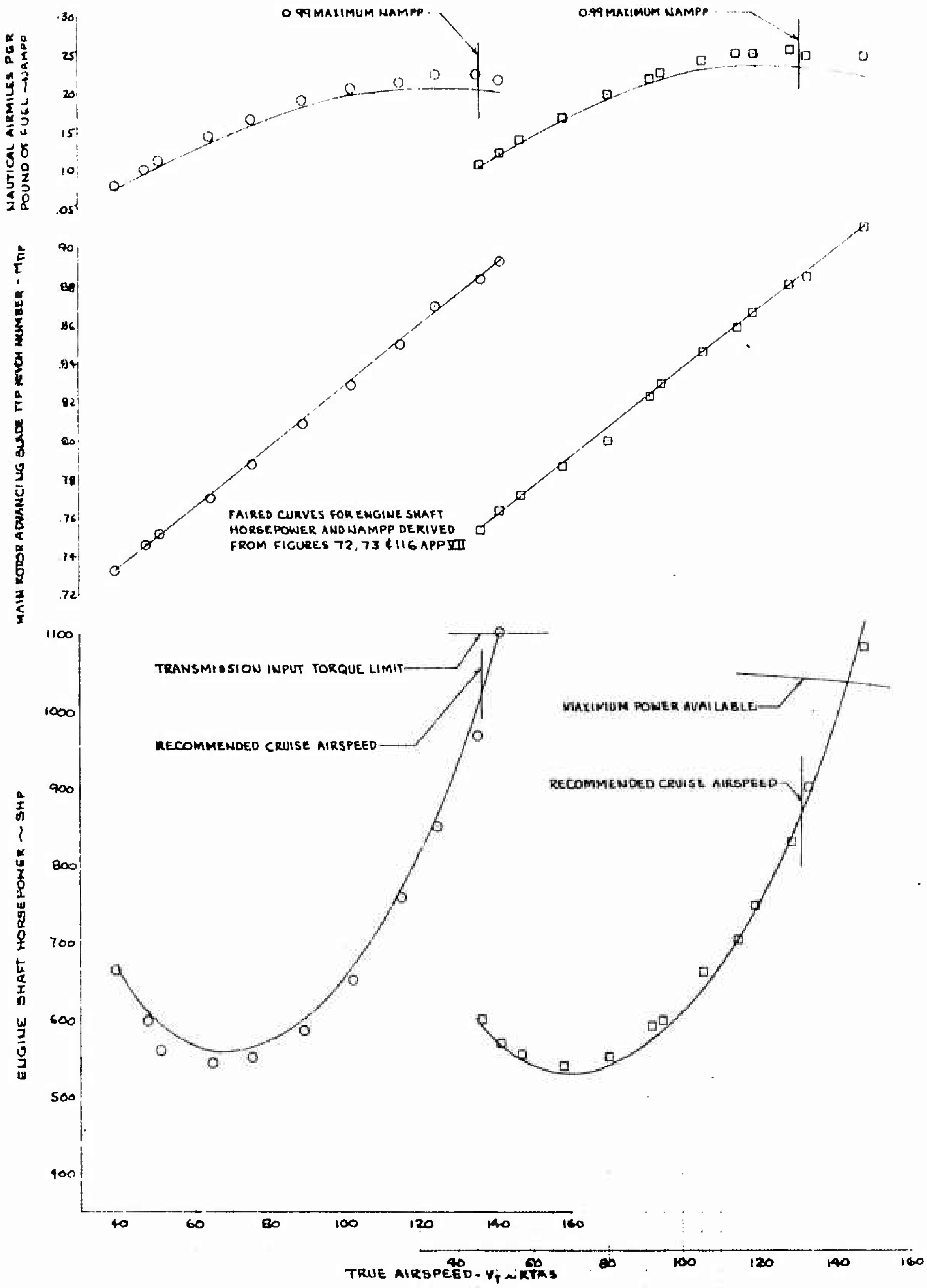


FIGURE NO 76

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

LT HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY - FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 77 APP VII

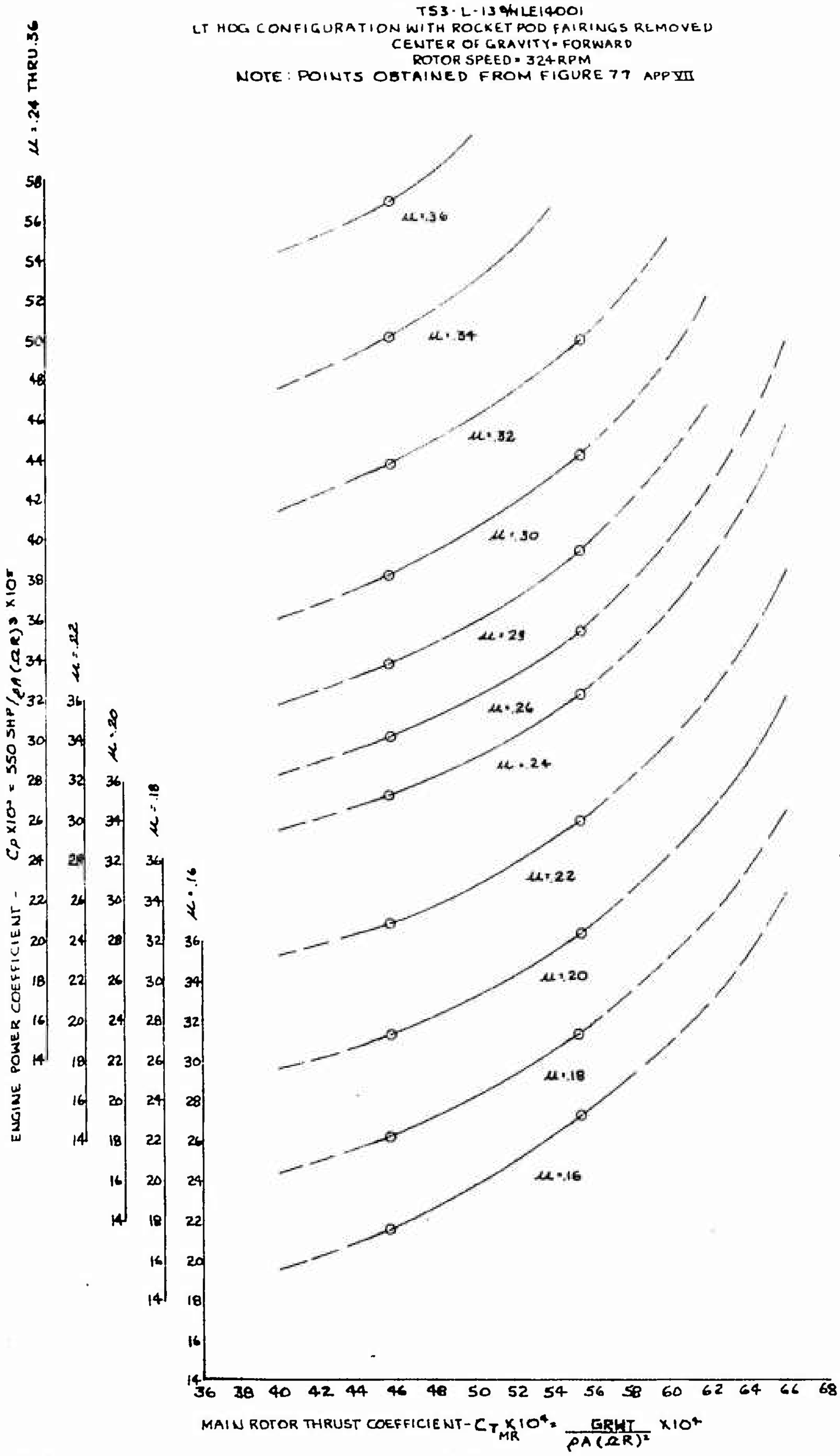


FIGURE NO 77

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	AVG. ALTITUDE H_D ~ FEET	AVG. GROSS WEIGHT ~ LB	AVG. LONG. C.G. ~ IN.	AVG. THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	3400	8260	190.9 (FWD)	0.004565	323	LT HOG
□	9320	8330	191.0 (FWD)	0.005548	322	LT HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

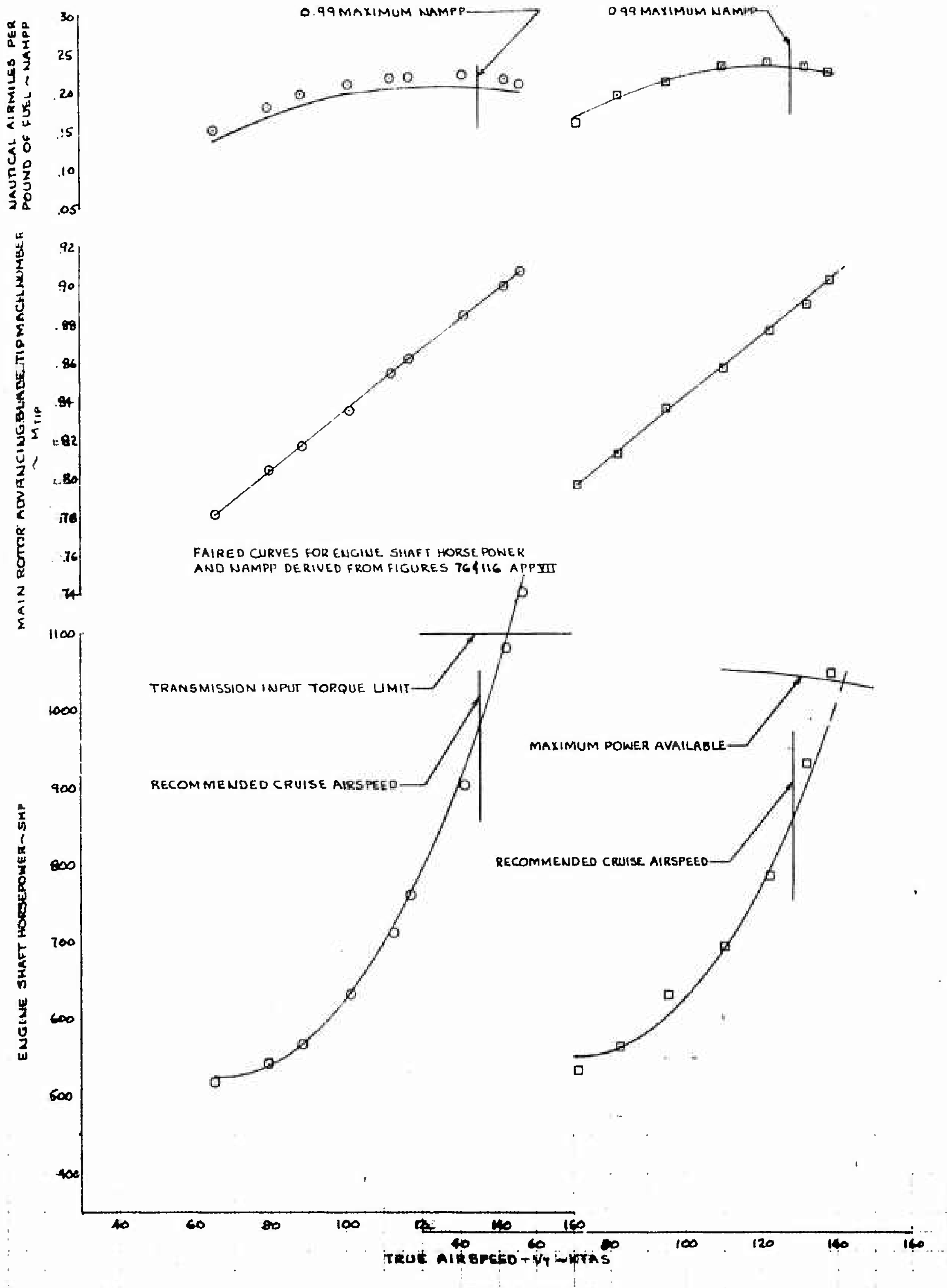


FIGURE NO. 78

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY. SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY = FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 80 THROUGH 83 APP. VII

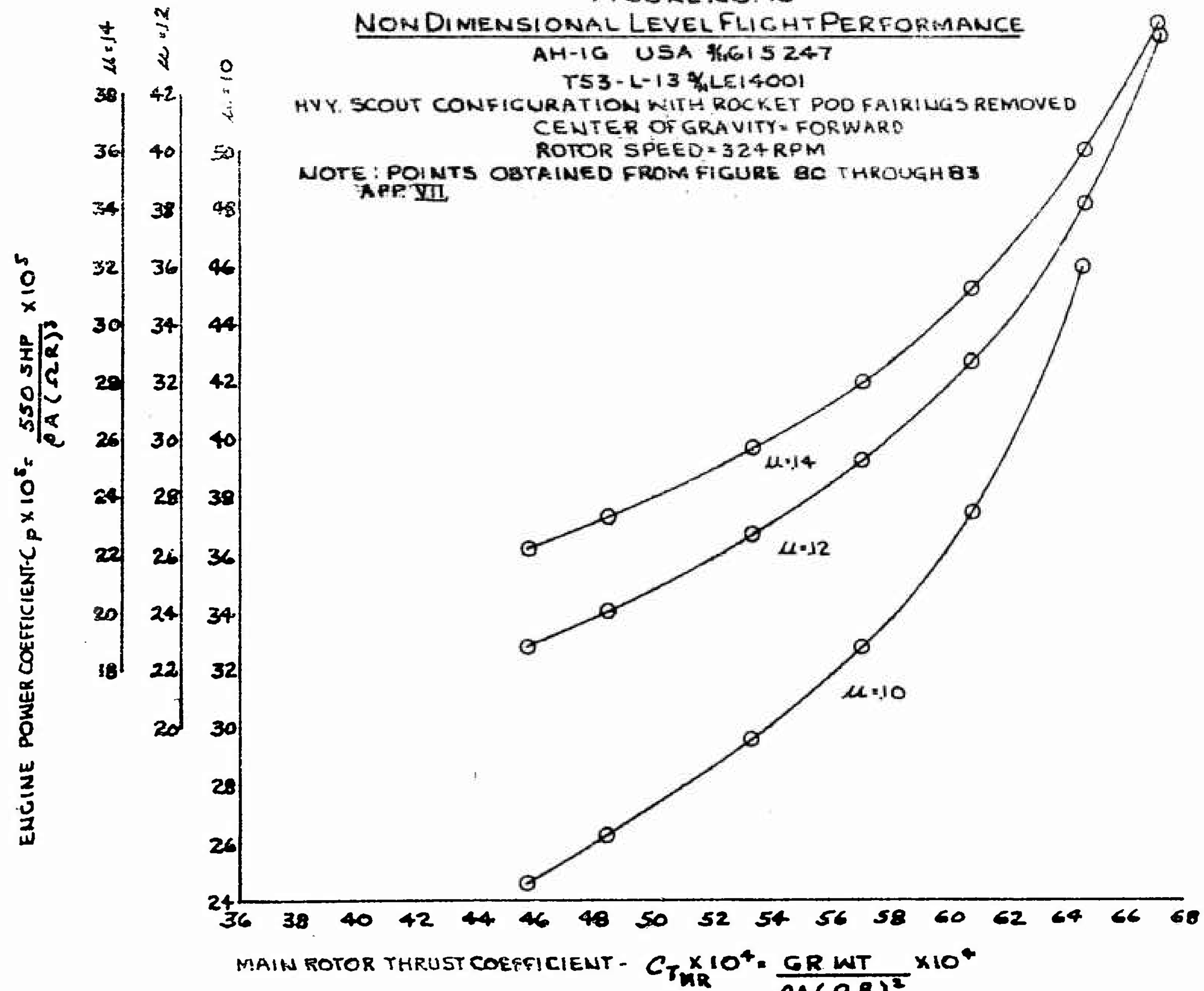


FIGURE NO 79

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY SCOUT CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY: FORWARD

ROTOR SPEED: 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 80 THROUGH 85 APP VII

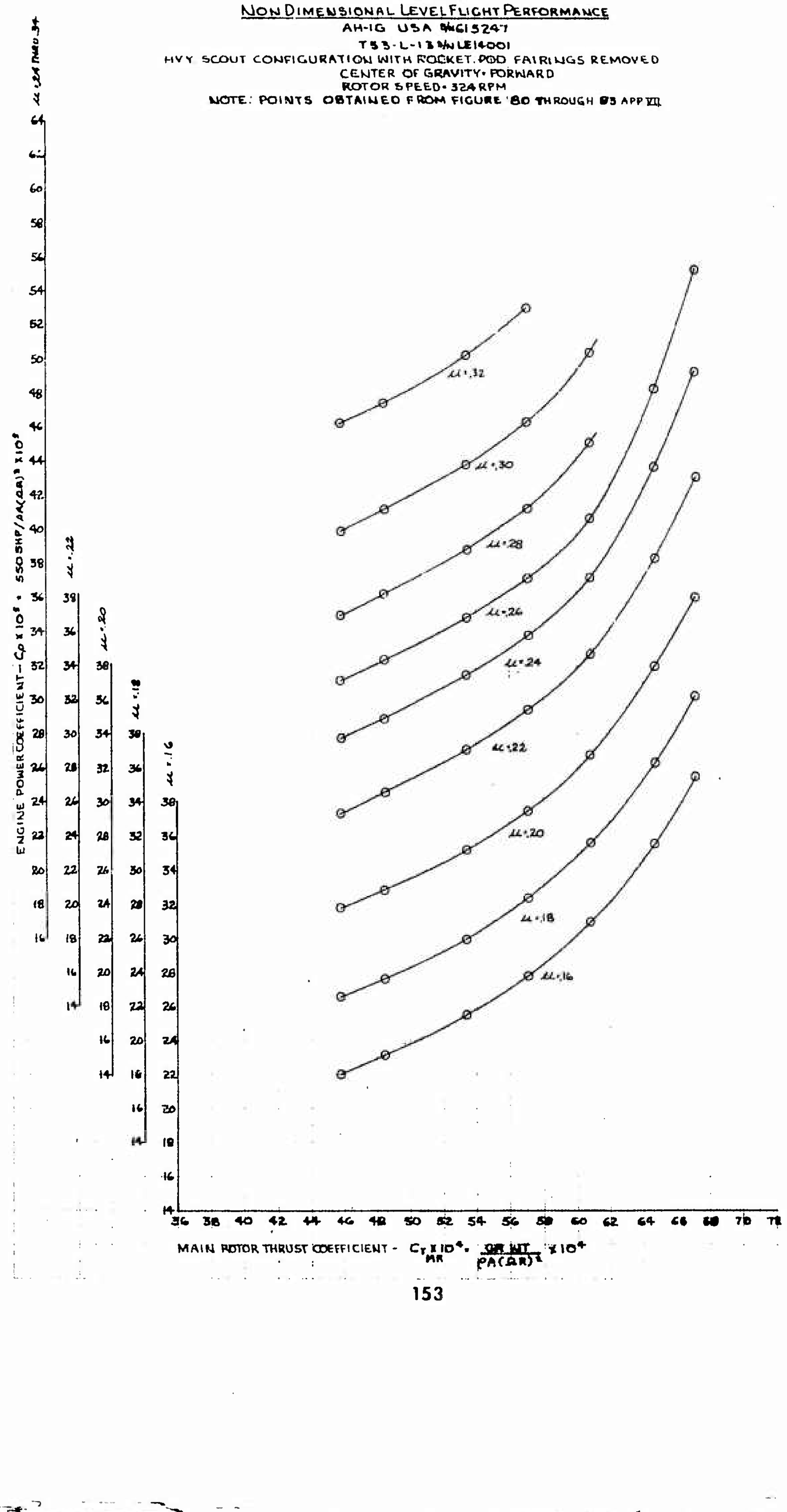


FIGURE NO 80
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYM	AVG. ALT H_D ~ FT	AVG GRWT ~ LB	AVG LONG CG ~ IN.	AVG THRUST COEFF. ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	4160	8170	193.3 (FWD)	0.004576	324.5	HVY. SCOUT
□	4310	8650	192.0 (FWD)	0.004852	325	HVY. SCOUT

NOTE: ALL ROCKET POD FAIRINGS REMOVED

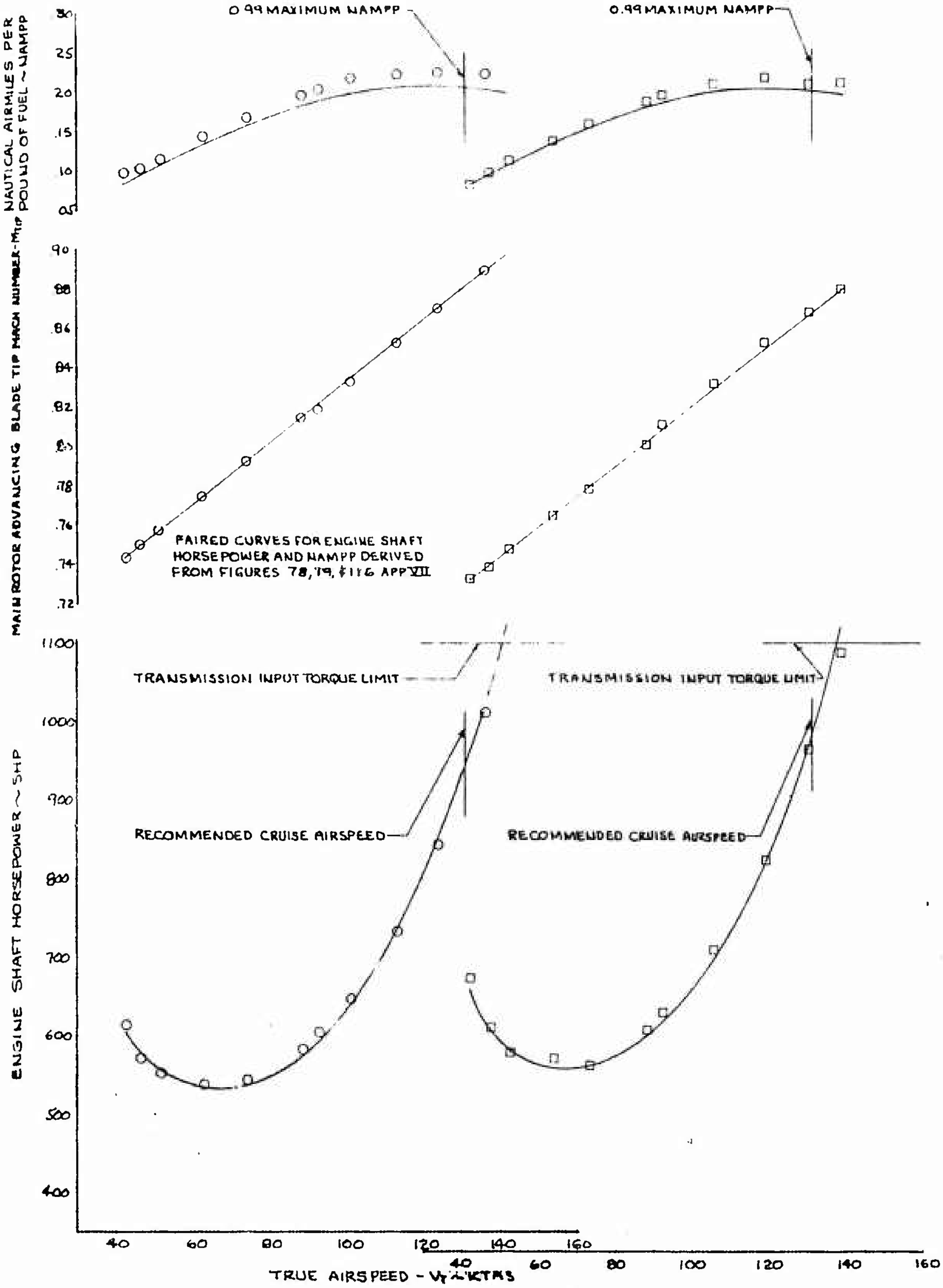


FIGURE NO 81

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG. ALT. H_D~FT	AVG GROSS WEIGHT ~LB	AVG. LONG CG ~IN.	AVG THRUST COEFF ~C_T	ROTOR SPEED ~RPM	ARMAMENT CONFIGURATION
○	8980	8250	193.3 (FWD)	0.005340	325.	HVY. SCOUT
□	9510	8620	191.5 (FWD)	0.005710	324.	HVY. SCOUT

NOTE: ALL ROCKET POD FAIRINGS REMOVED

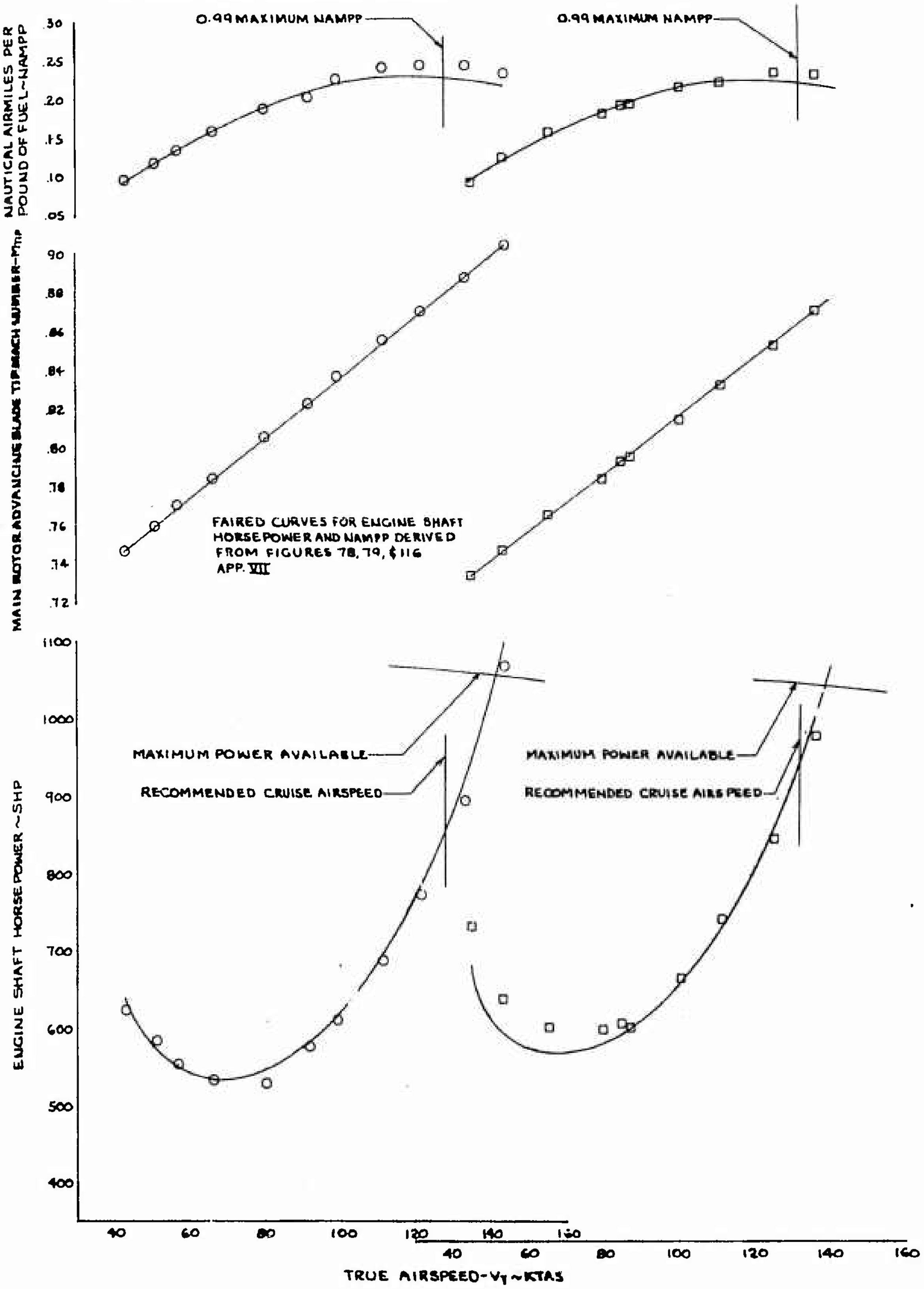


FIGURE No 82

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALT H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG CG ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	9440	9120	191.9 (FWD)	0.006080	522.5	HVY SCOUT
□	14650	8240	191.2 (FWD)	0.006469	323	HVY SCOUT

NOTE: ALL ROCKET POD FAIRINGS REMOVED

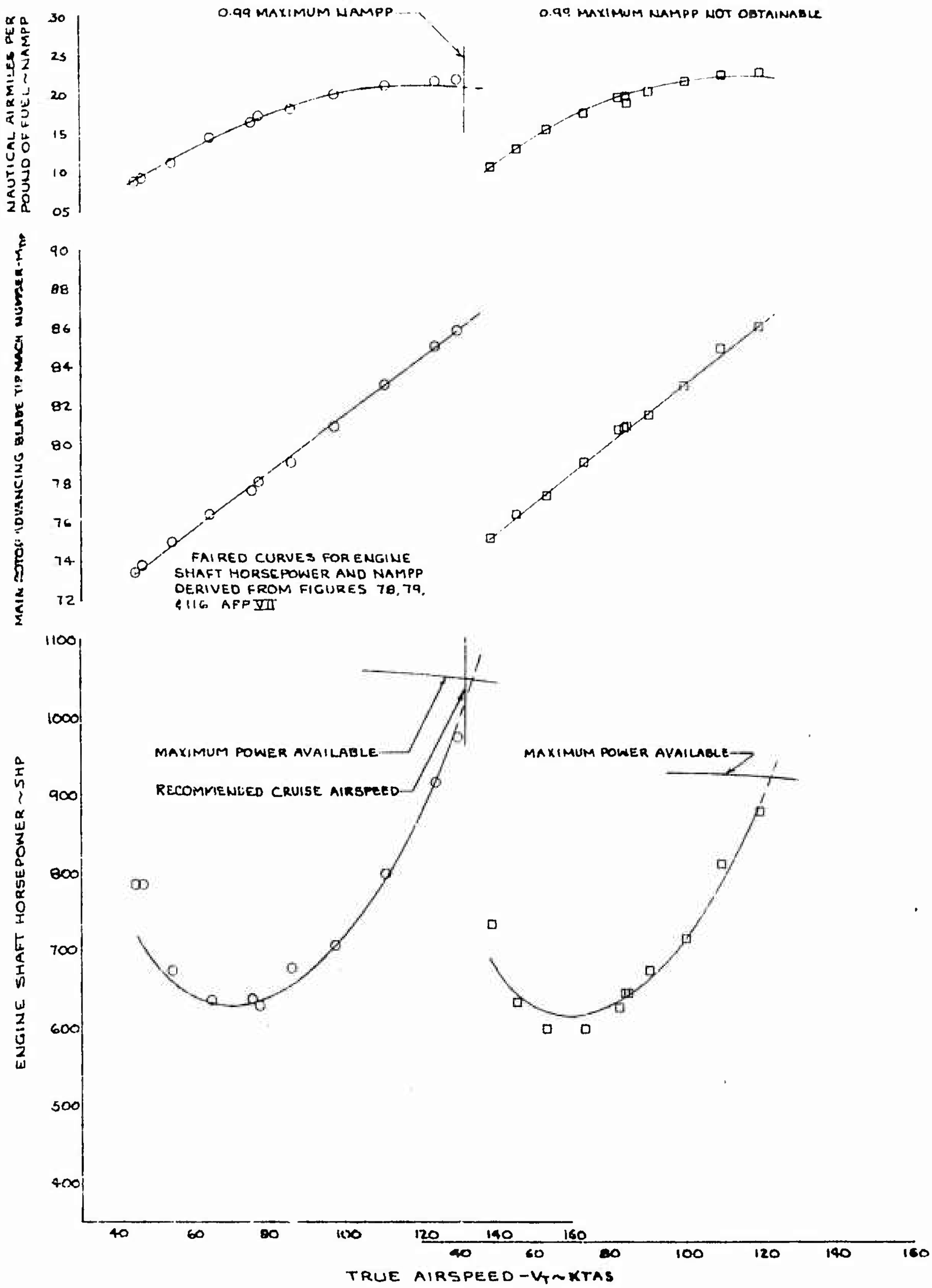


FIGURE NO 83

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53 L-13 S/N LE14001

SYM	AVG ALT H_O ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG CG ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIGURATION
○	14410	8650	191.3 (FWD)	0.006717	323.5	HVY SCOUT

NOTE: ALL ROCKET POD FAIRINGS REMOVED

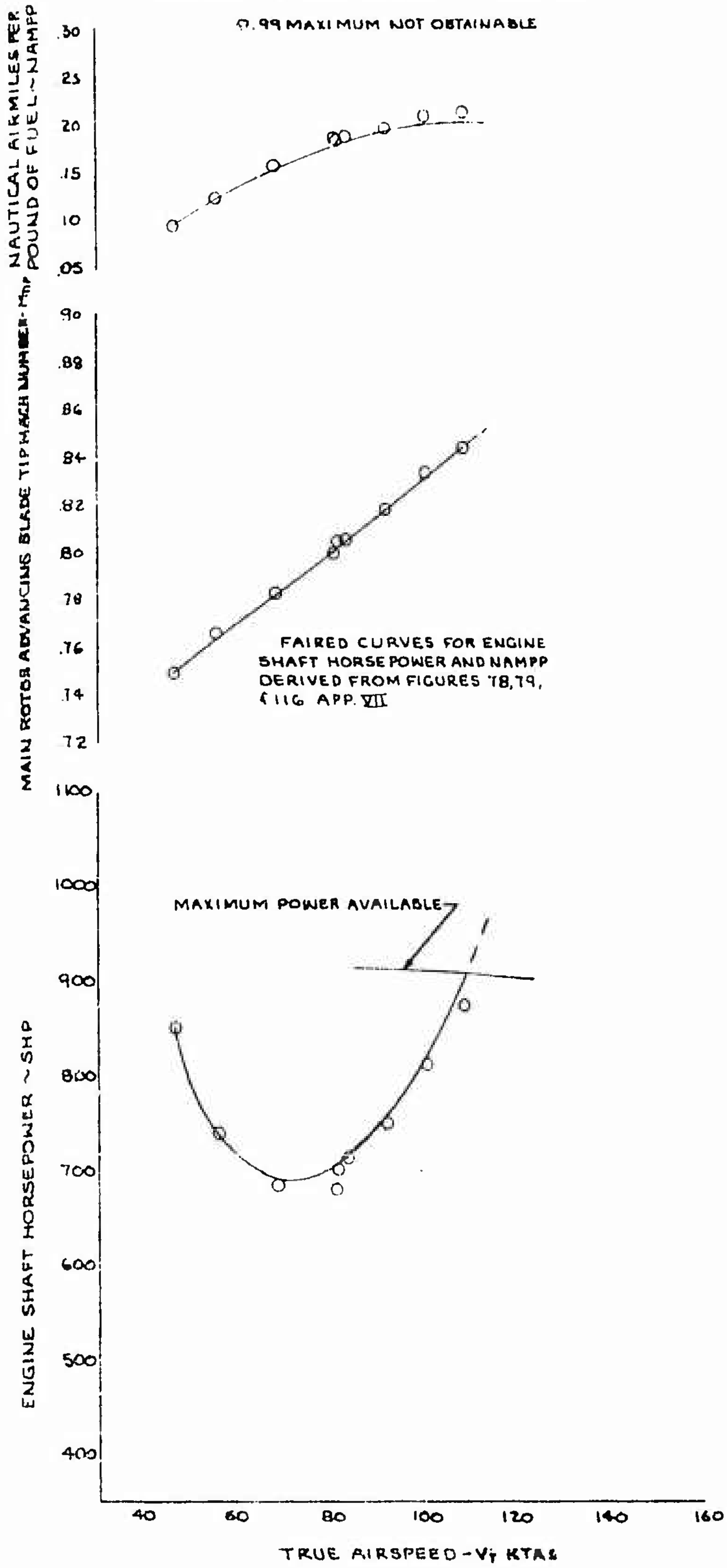


Figure No 84

Non Dimensional Level Flight Performance

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY = FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 86 THROUGH 91 APP VII

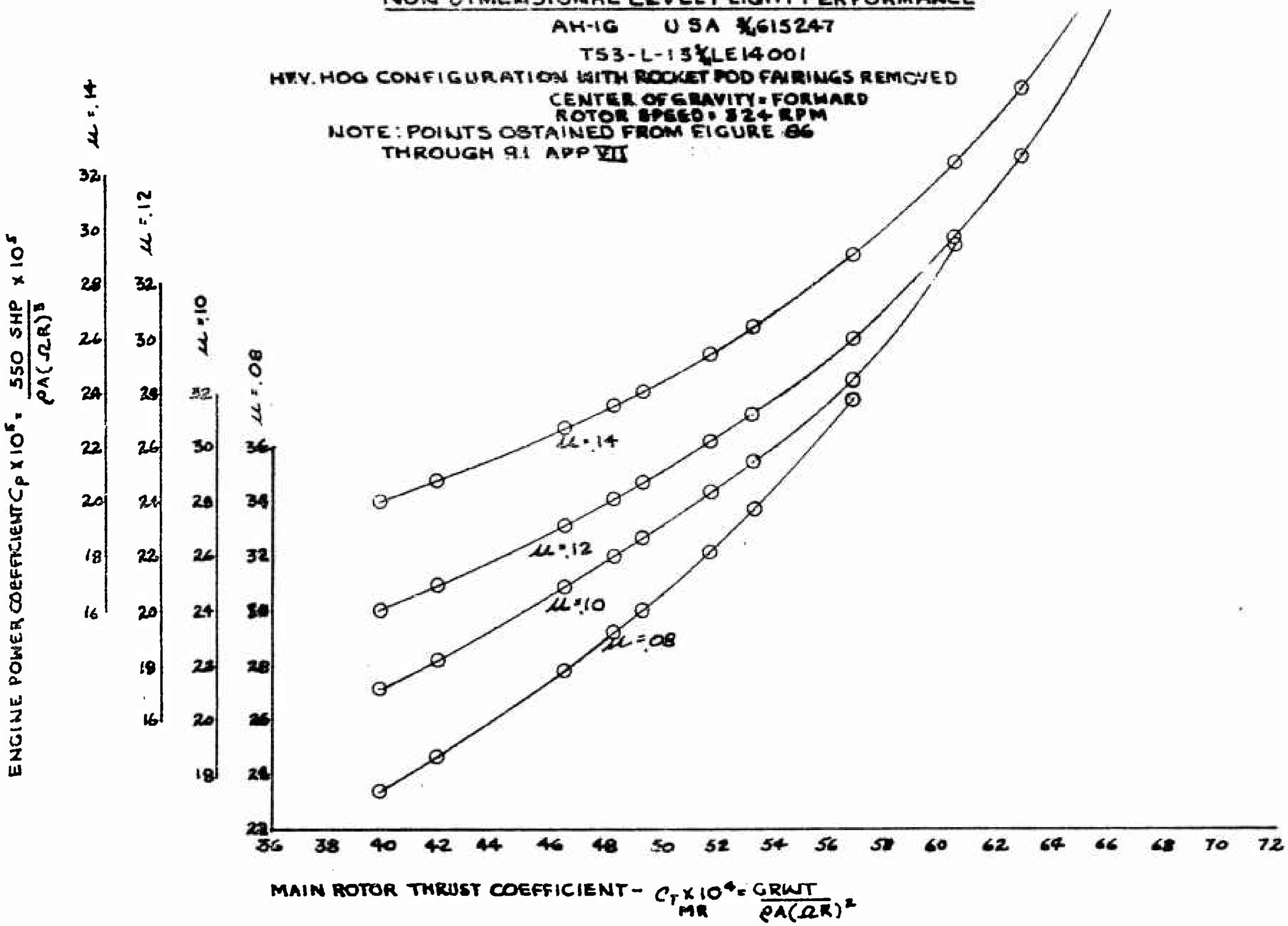


FIGURE NO. 85

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY = FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 86 THROUGH 91 APP VII

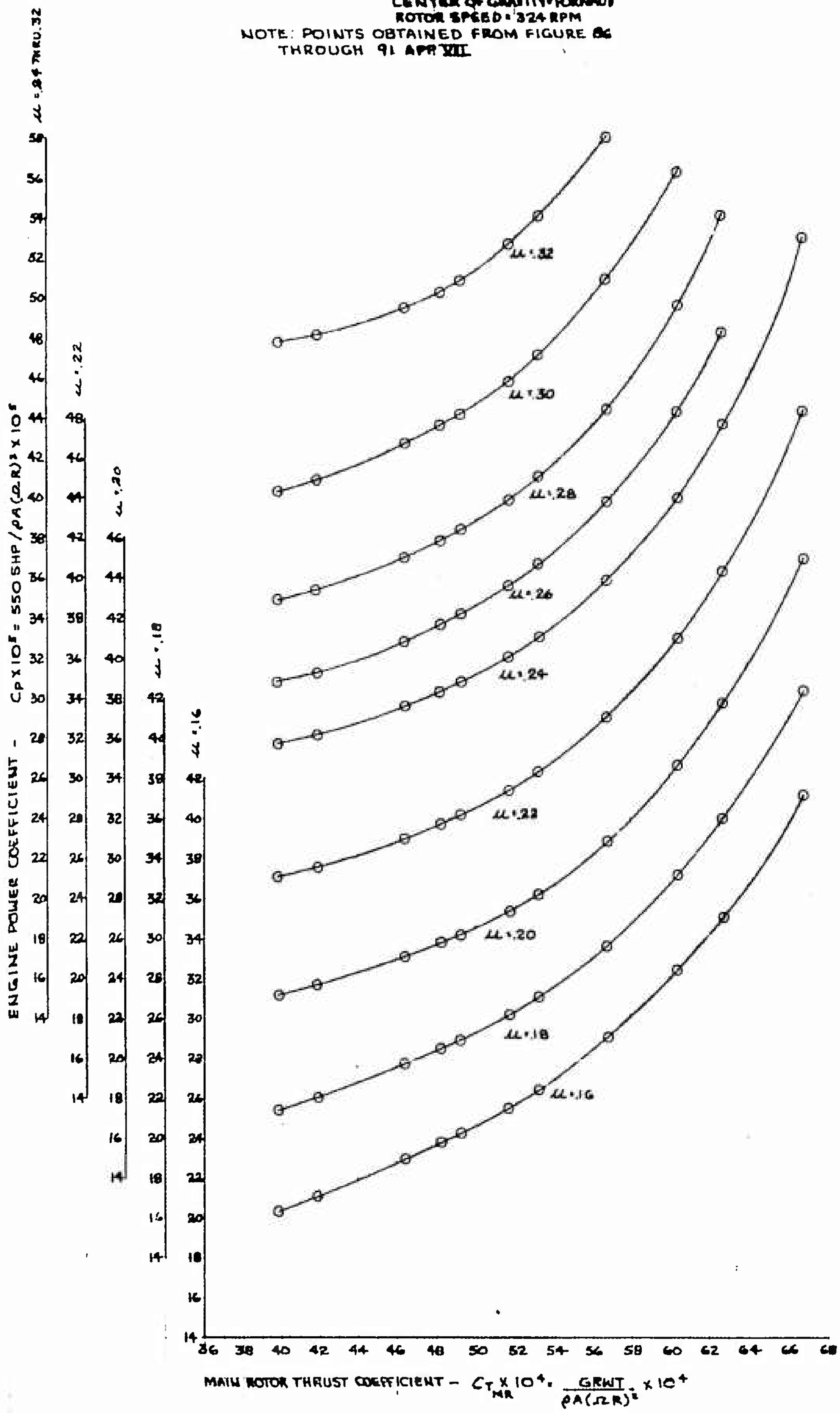


FIGURE NO. 86

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG. GROSS WEIGHT ~ LB.	AVG. LONG CG ~ IN.	AVG. THRUST COEFF ~C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	1490	7680	191.1 (FWD.)	0.003988	324	HVY. HOG
□	3470	7620	190.3 (FWD.)	0.004195	324	HVY. HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

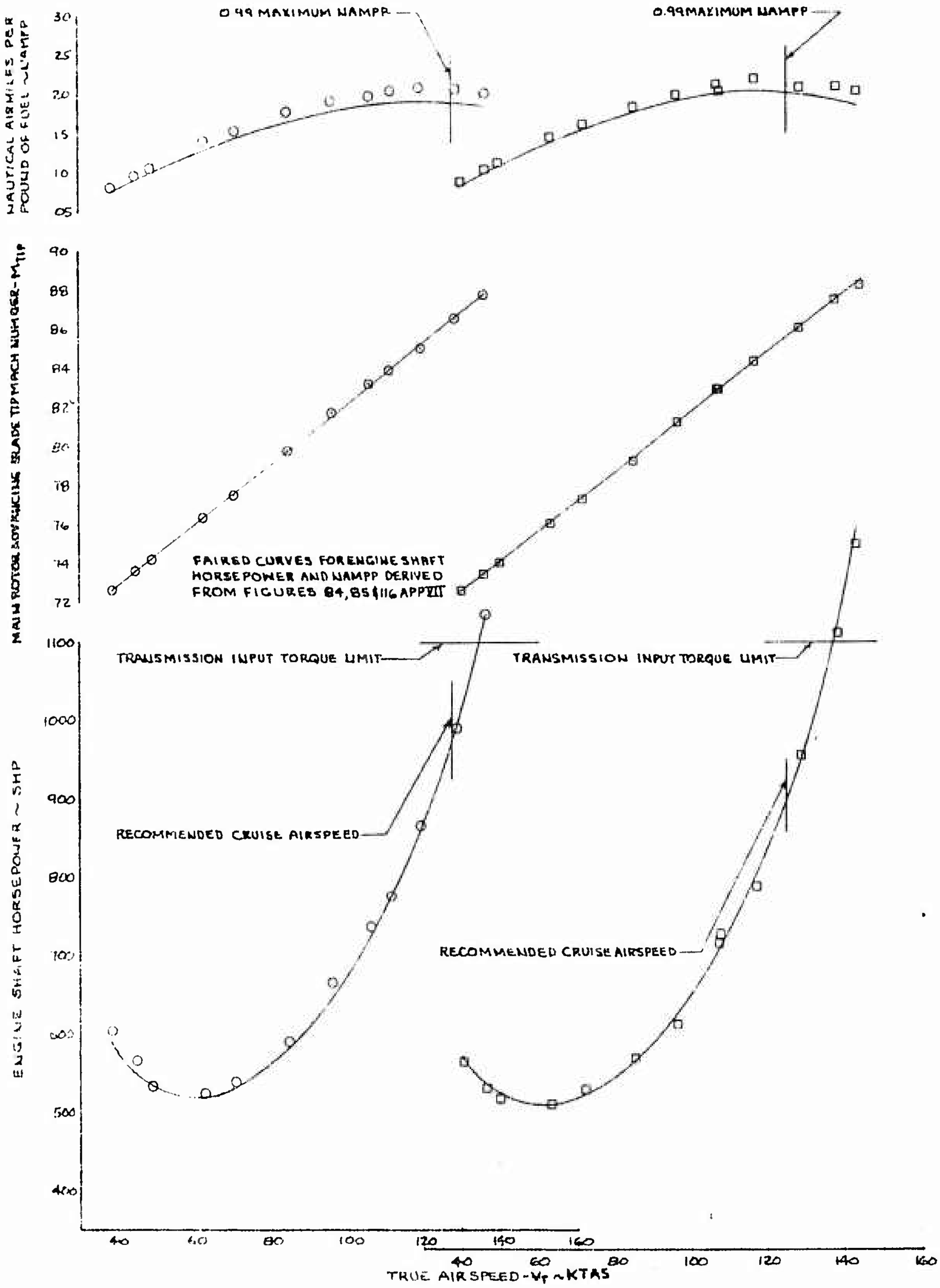


FIGURE NO 87

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG CG ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	4180	8320	192.1 (FWD)	0.004693	325	HVY. HOG
□	4290	8630	191.6 (FWD)	0.004823	325.5	HVY. HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

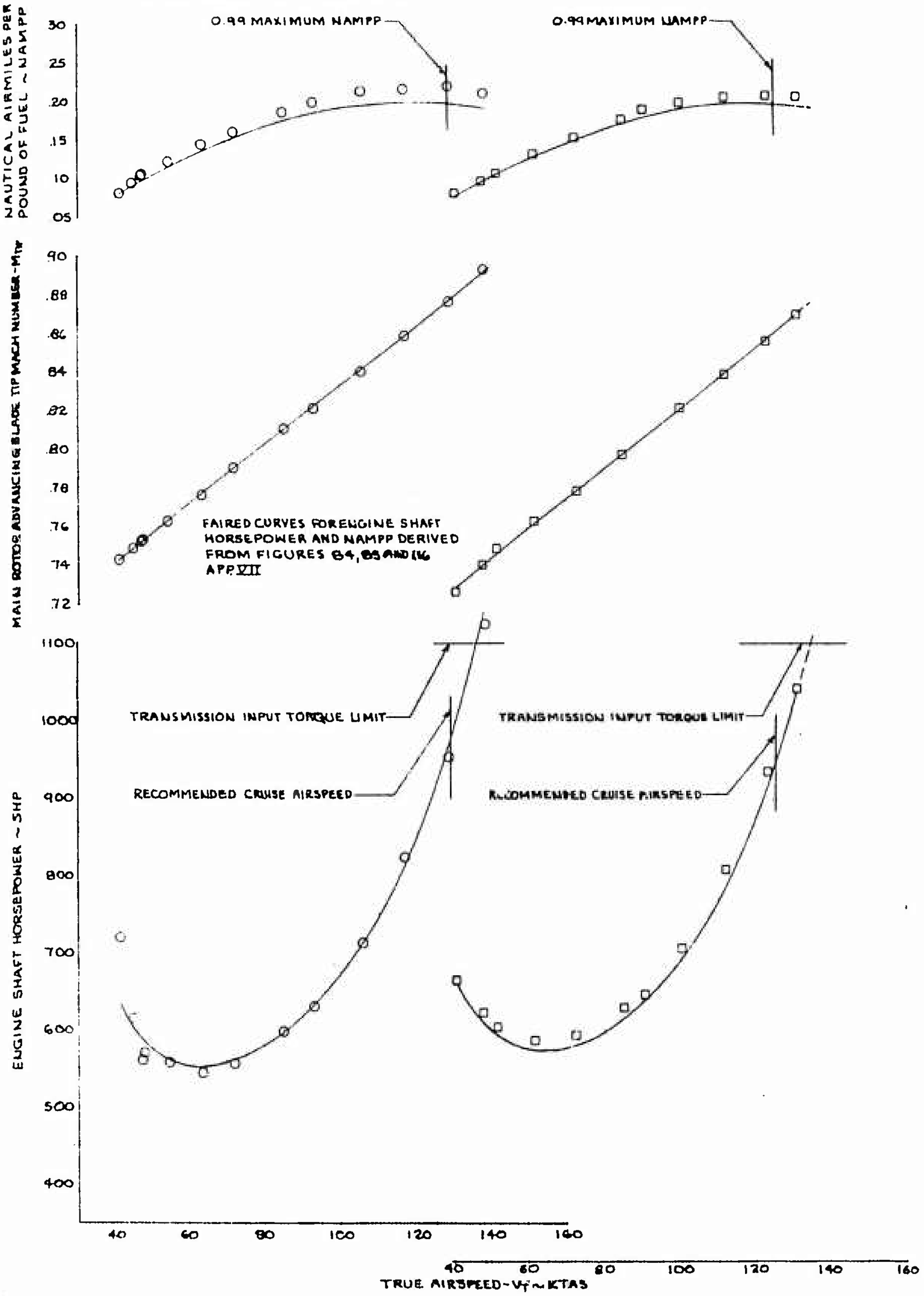


FIGURE NO 88

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG. ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG. LONG CG ~ IN.	AVG. THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG.
○	3920	8830	191.8 (FWD)	0.004925	324	HVY. HOG
□	4190	9210	193.1 (FWD)	0.005163	324.5	HVY HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

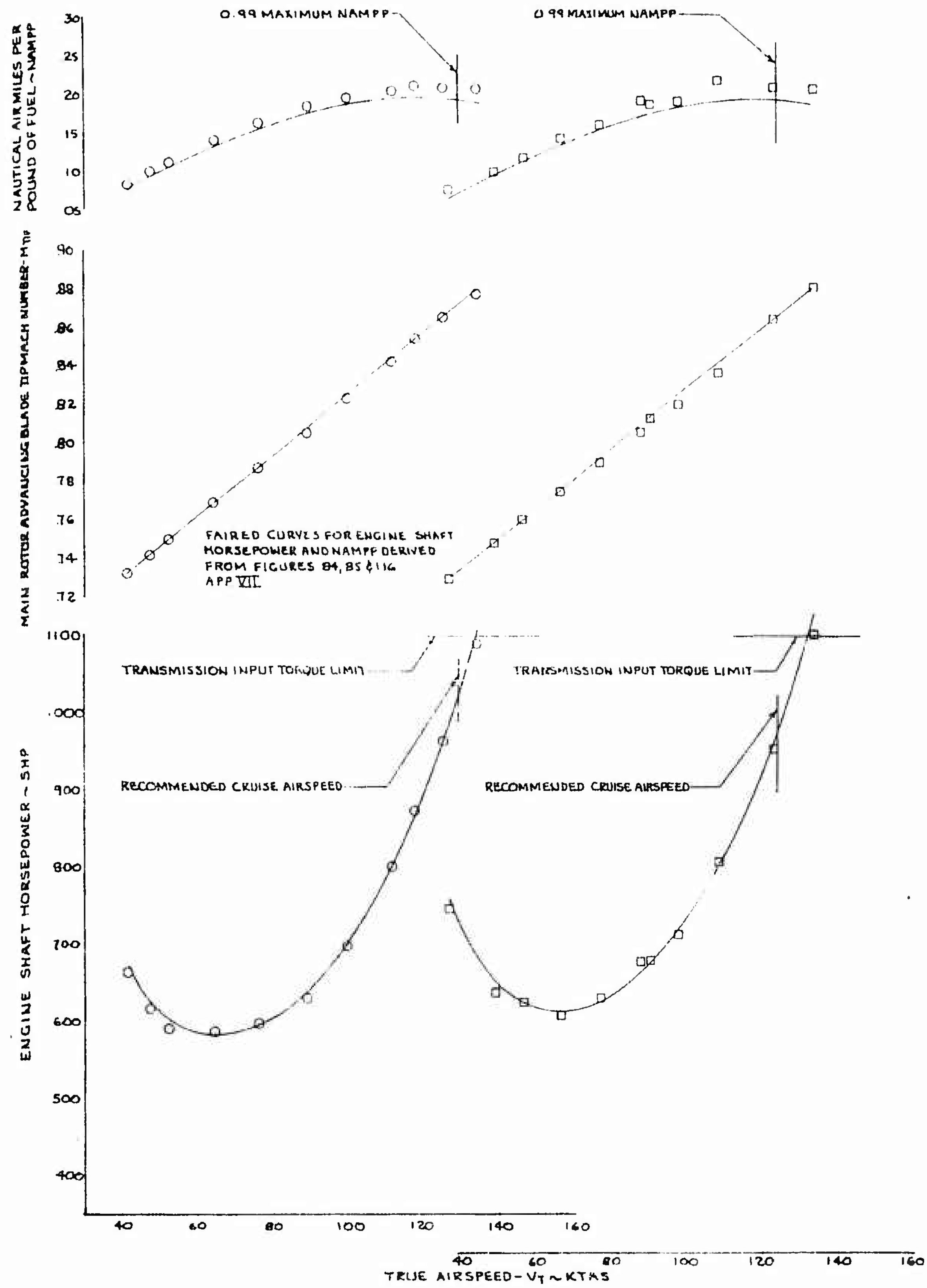


FIGURE NO 89

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG CG ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	9410	8070	192.6	0.005320	324.5	HVY HOG
□	9390	8640	191.6	0.005682	324.5	HVY HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

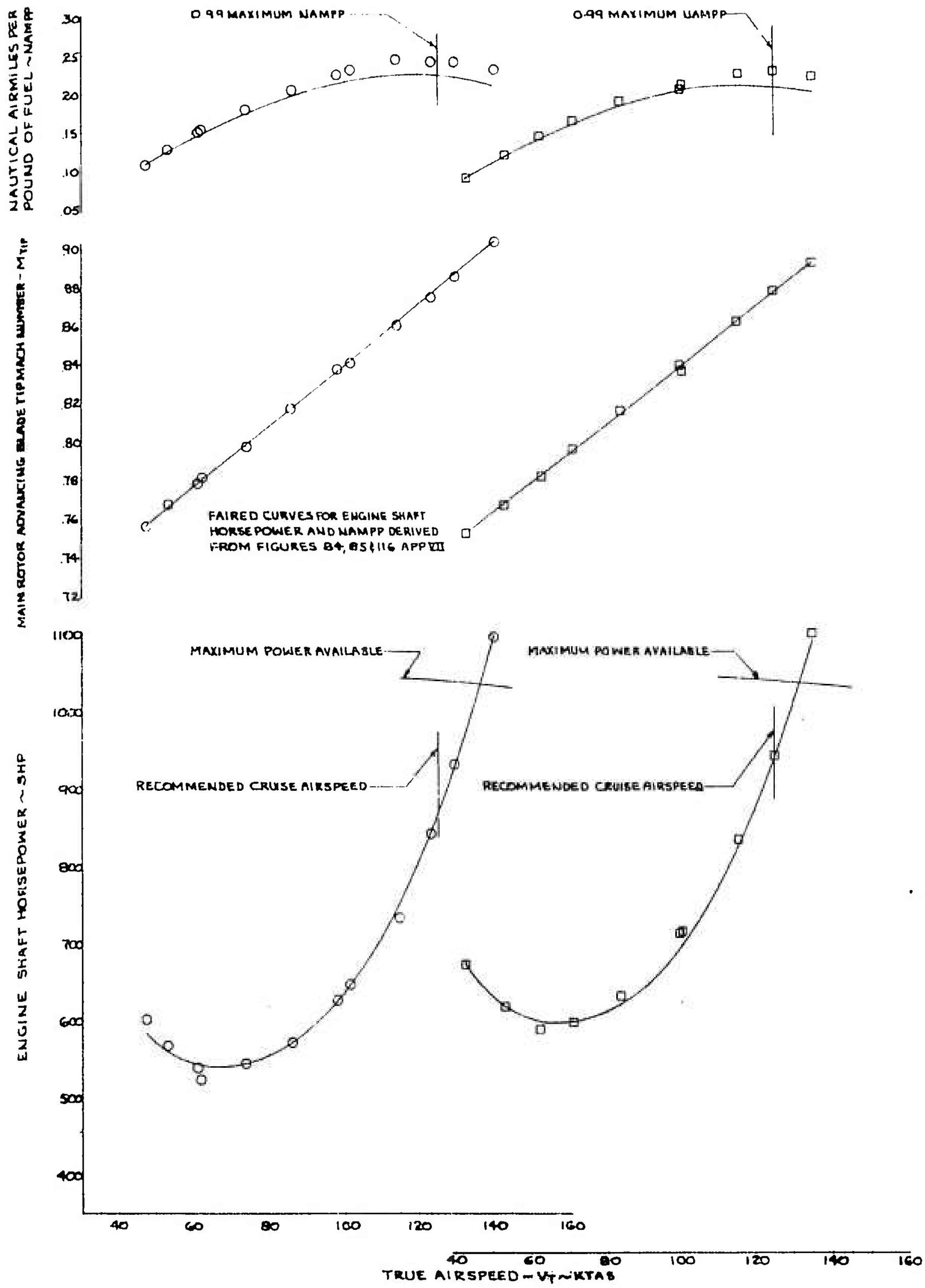


FIGURE NO 90

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 66-15247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_p ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG CG ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	9190	9190	192.5(FWD)	0.006044	323.5	HVY HOG
□	14410	8070	191.3(FWD)	0.006279	323.5	HVY HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

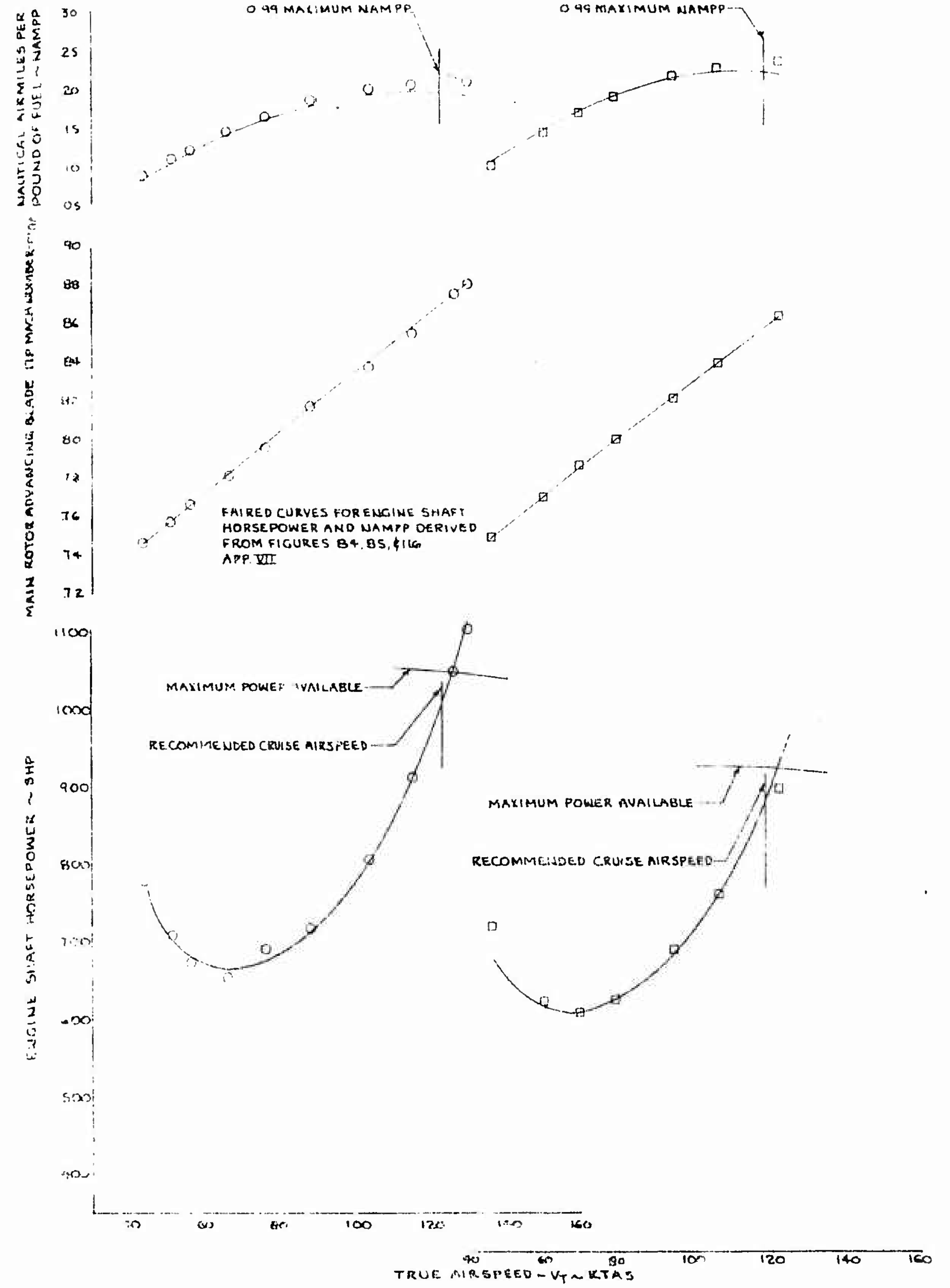


FIGURE NO 91

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 66-15247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~FT	AVG GROSS WEIGHT ~LB	AVG LONG CG ~IN	AVG THRUST COEFF ~C_T	ROTOR SPEED ~RPM	ARMAMENT CONFIG
○	14410	8570	192.1 (FWD)	0.006676	323	HVY HOG

NOTE ALL ROCKET POD FAIRINGS REMOVED

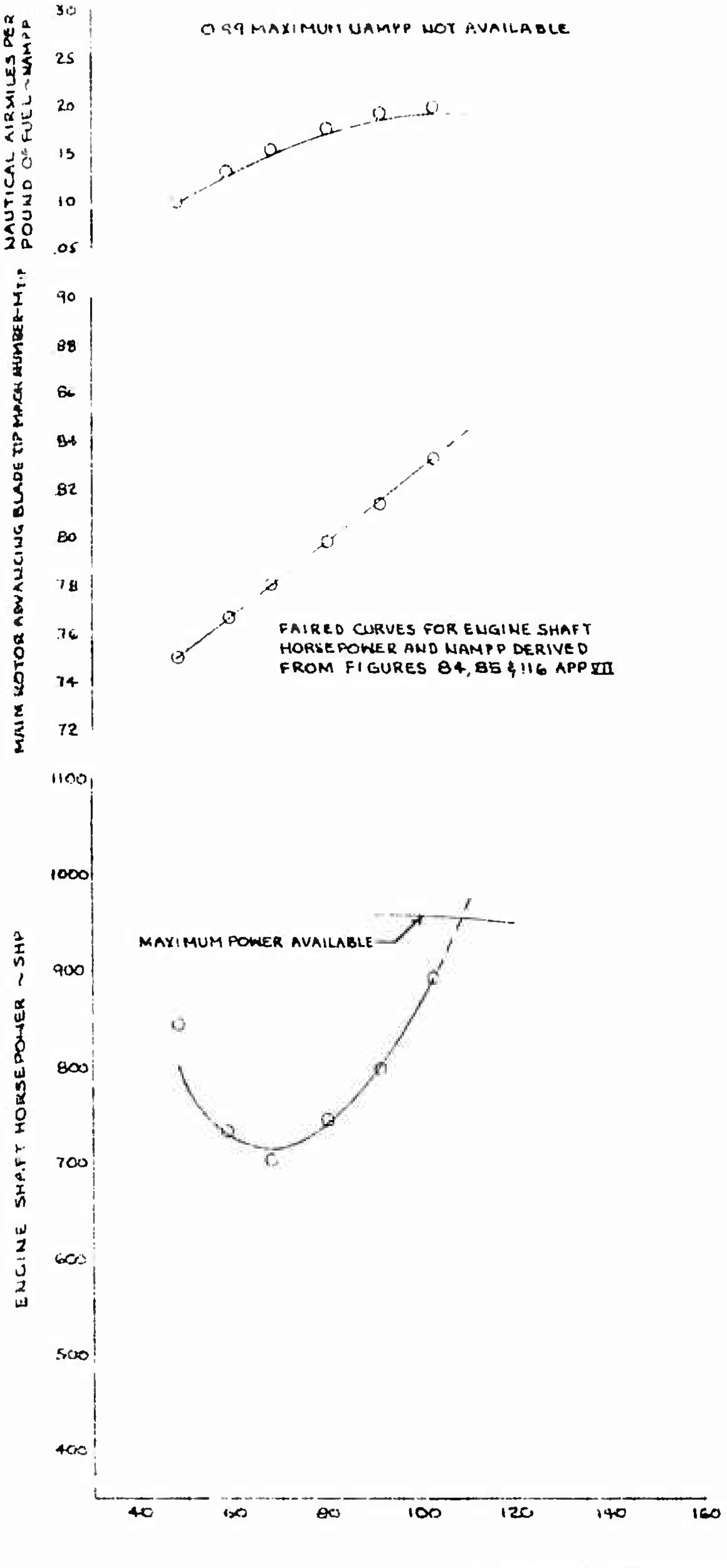


FIGURE NO. 92

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS INSTALLED

CENTER OF GRAVITY: FORWARD

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURES 94 & 95 APP. VII

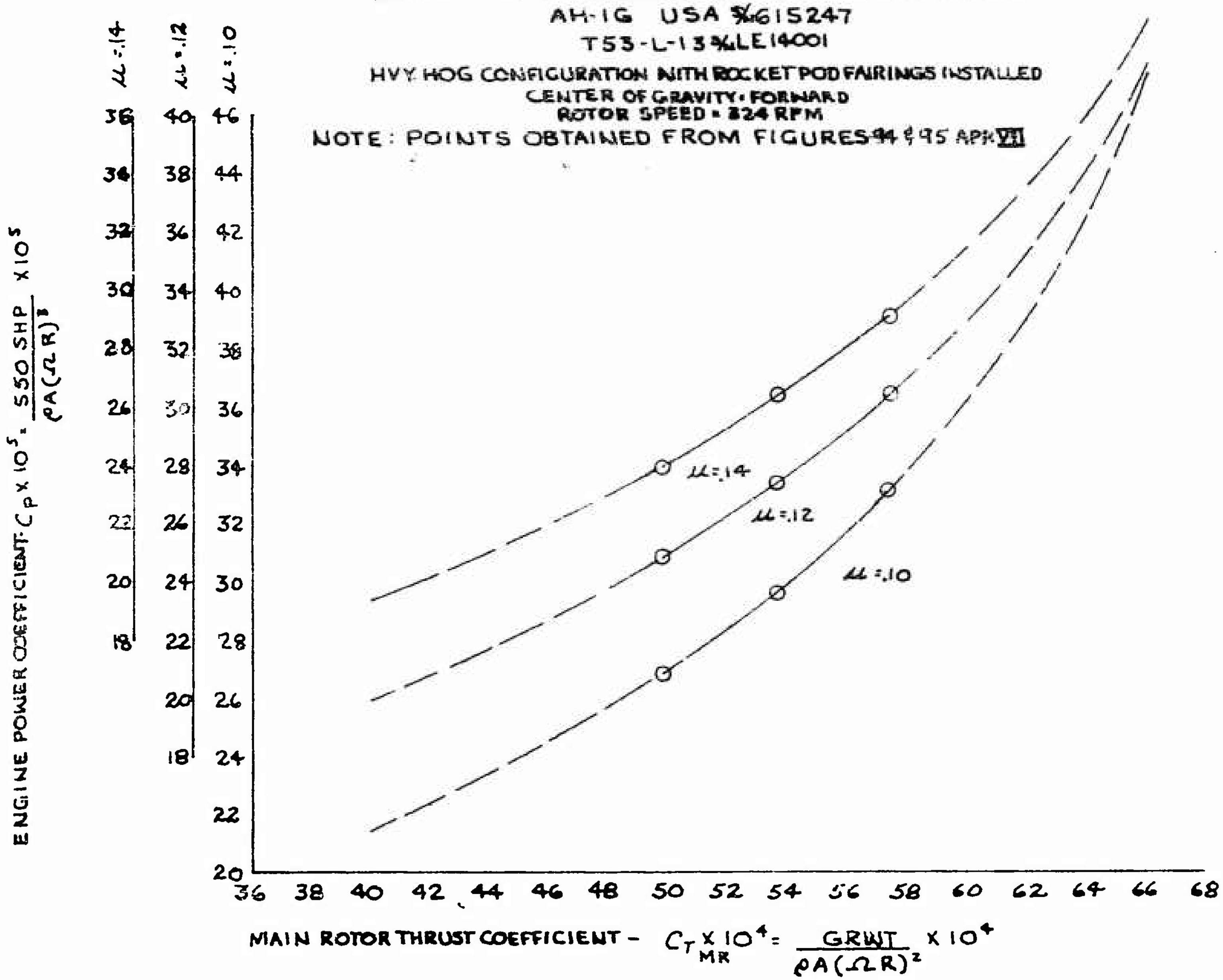


FIGURE NO 93

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS INSTALLED

CENTER OF GRAVITY: FORWARD

ROTOR SPEED: 324 RPM

NOTE POINTS OBTAINED FROM FIGURES 94 & 95 APP. VII

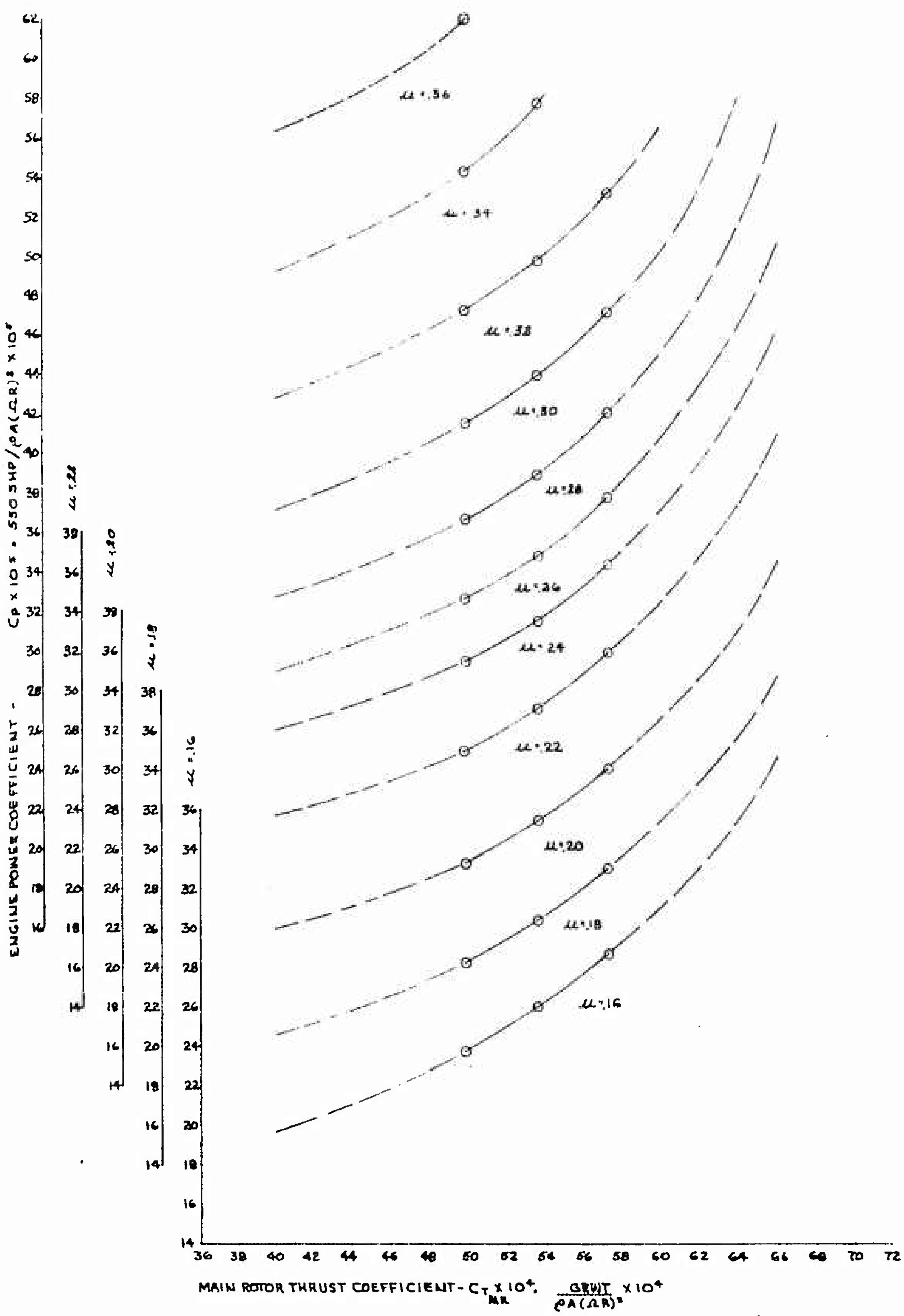


FIGURE NO 94

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LBS	AVG LONG C.G. ~ IN	AVG THRUST COEF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	4890	8560	192.1 (FWD)	0.004976	324	HVY HOG
□	8410	8350	192.2 (FWD)	0.005859	323.5	HVY HOG

NOTE: ALL ROCKET POD FAIRINGS INSTALLED

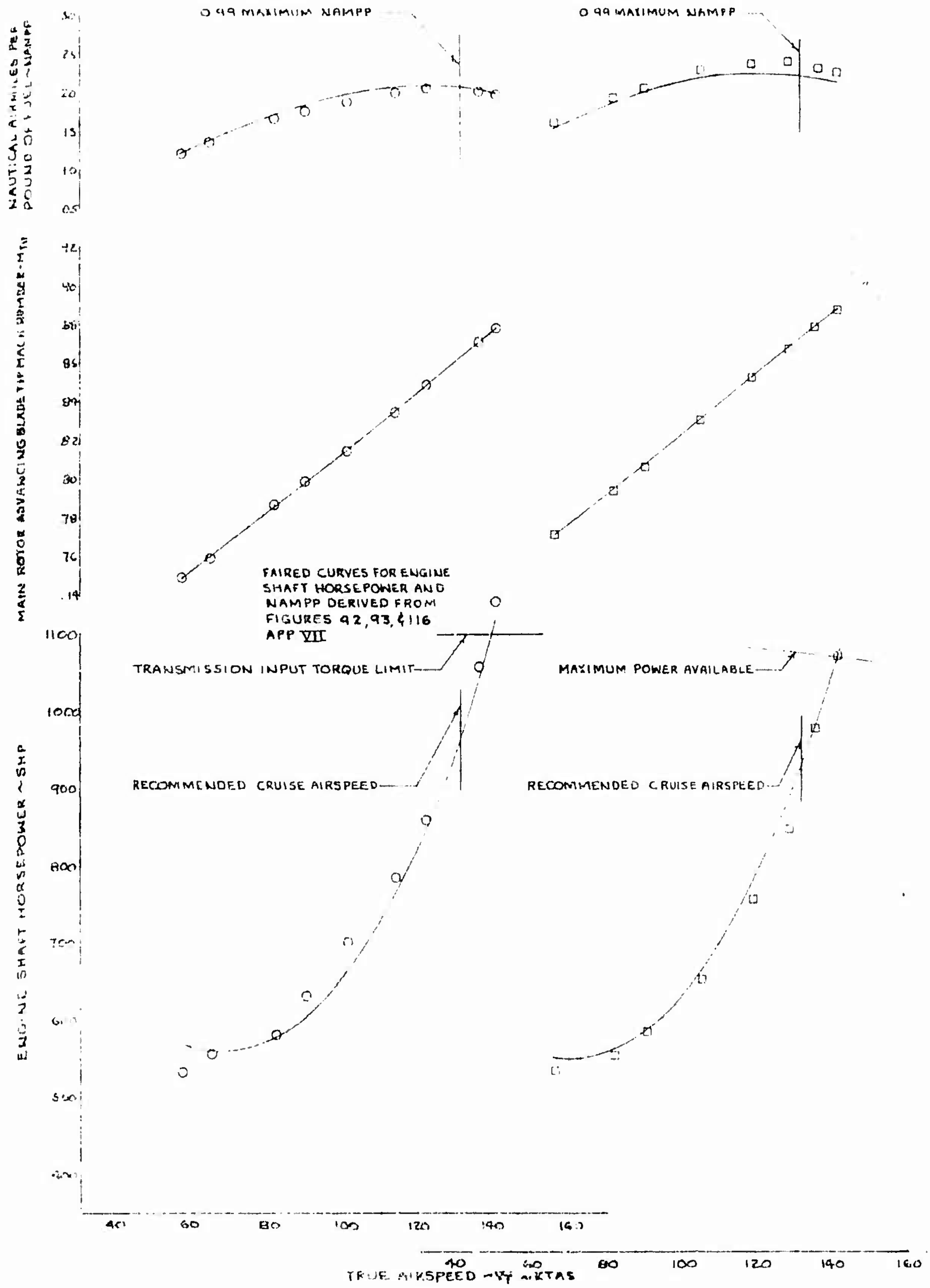


FIGURE NO 95

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LBS	AVG LONG CG ~ IN	AVG THRUST COEF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	9120	8740	192.4(FWD)	0.005755	323.5	HVY HOG

NOTE: ALL ROCKET POD FAIRINGS INSTALLED

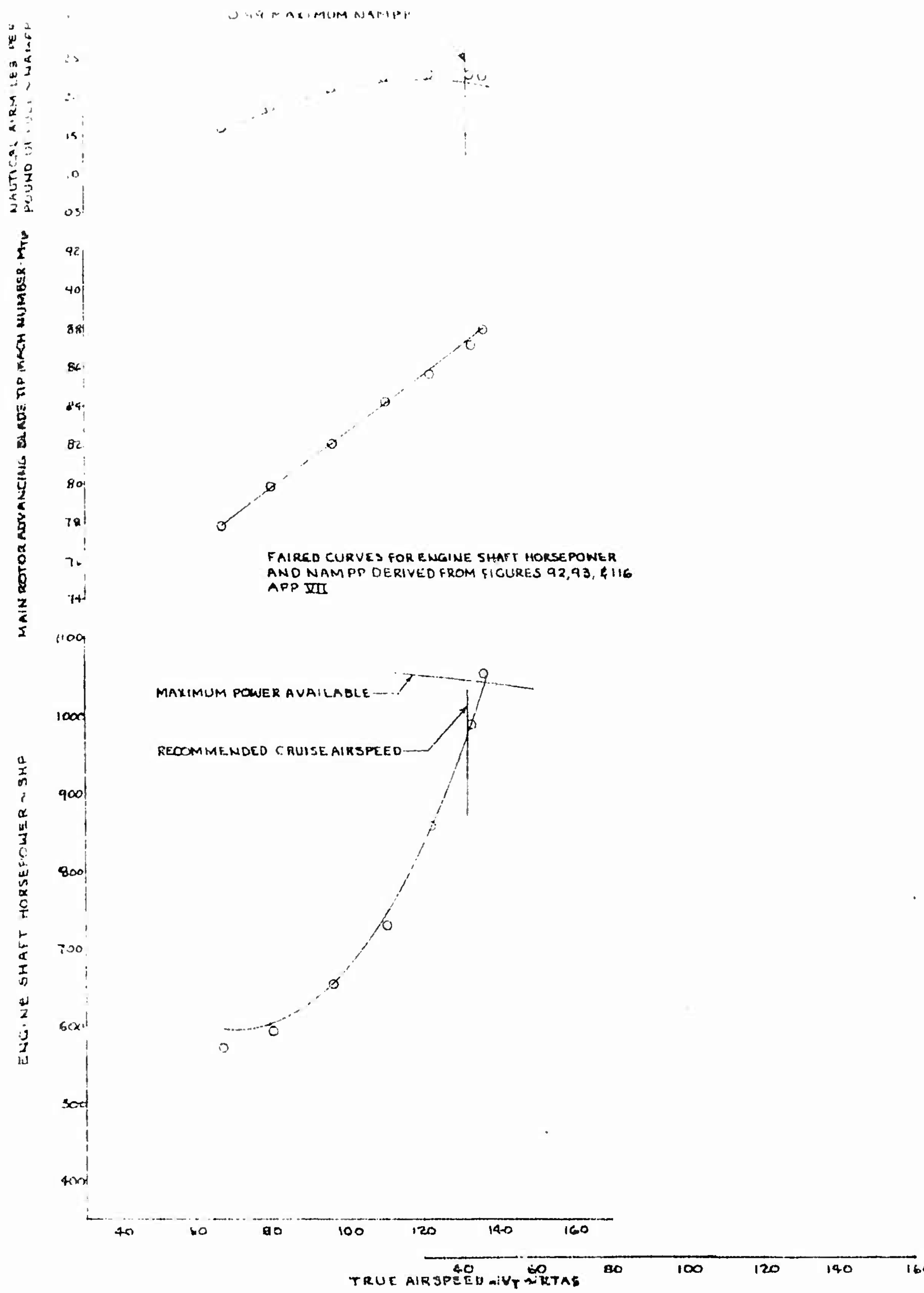


FIGURE No 96

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY: AFT

ROTOR SPEED = 324 RPM

NOTE: POINTS OBTAINED FROM FIGURE 98 THROUGH 101 APP VII

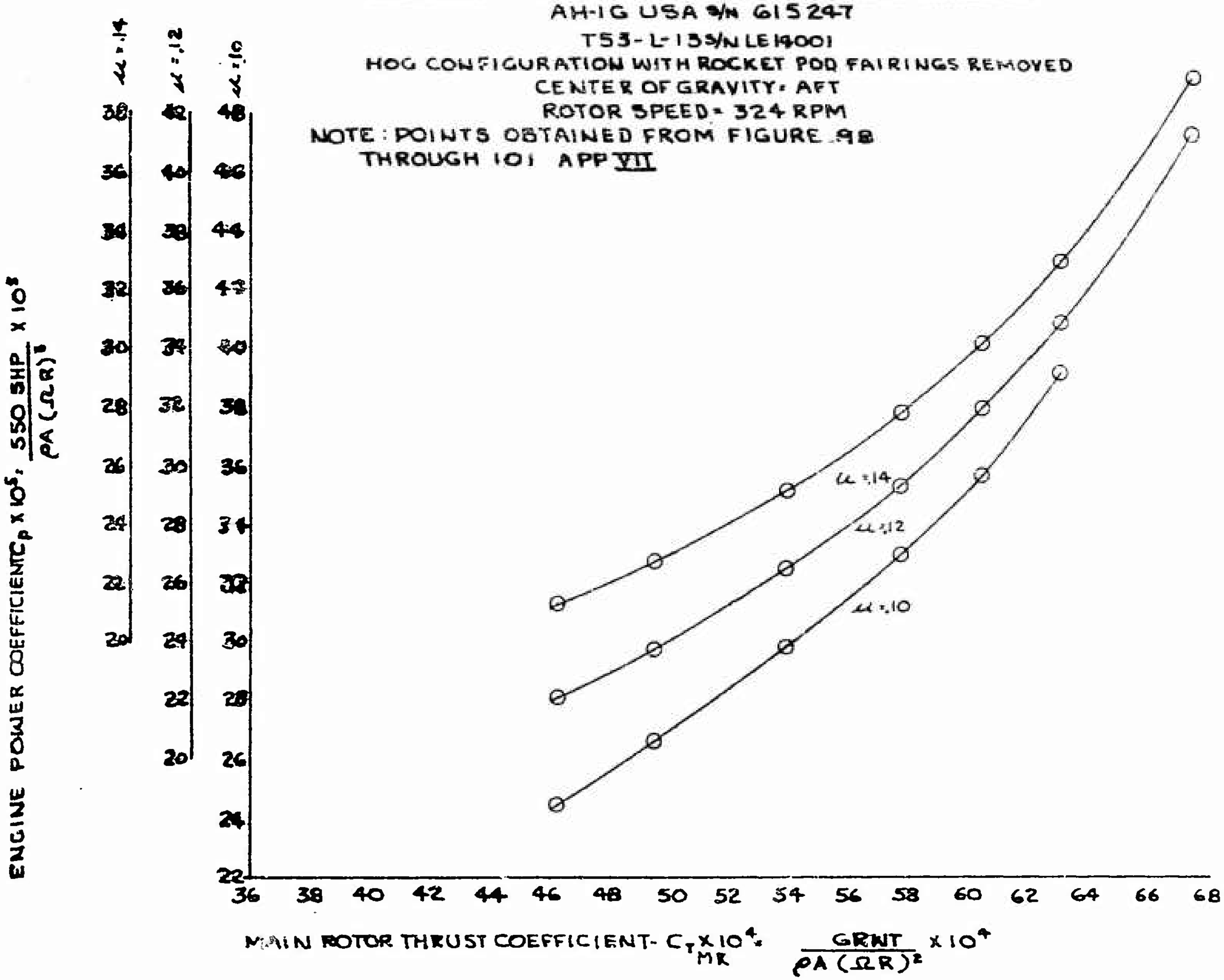


FIGURE No 97

NON DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14-001

HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY: AFT

ROTOR SPEED = 324 RPM

NOTE POINTS OBTAINED FROM FIGURE 98 THROUGH 101 APP. VII

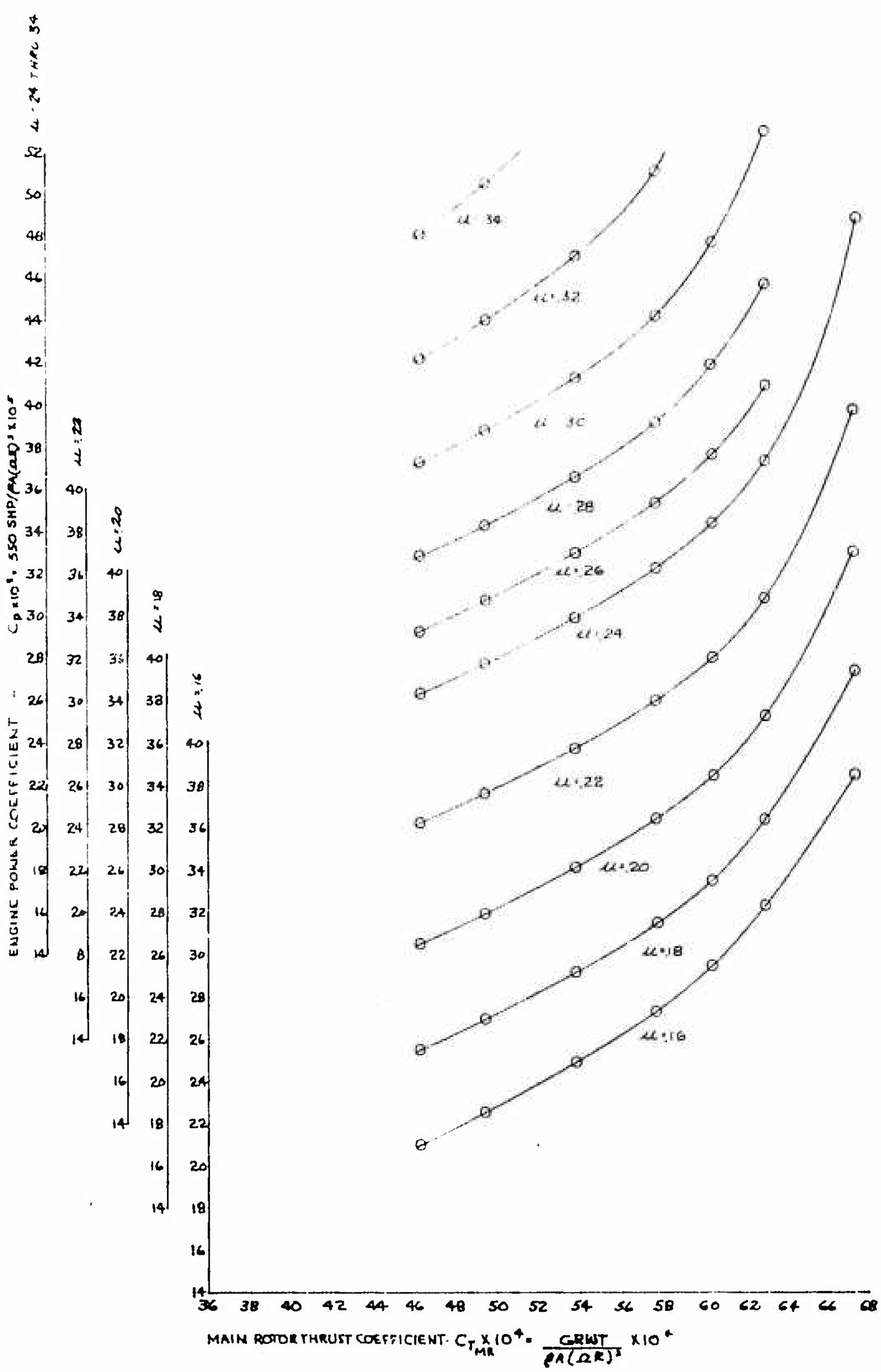


FIGURE NO 98
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	4910	8020	199.0 (AFT)	0.004624	323.5	HVY HOG
□	4760	8560	199.9 (AFT)	0.004946	322.5	HVY HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

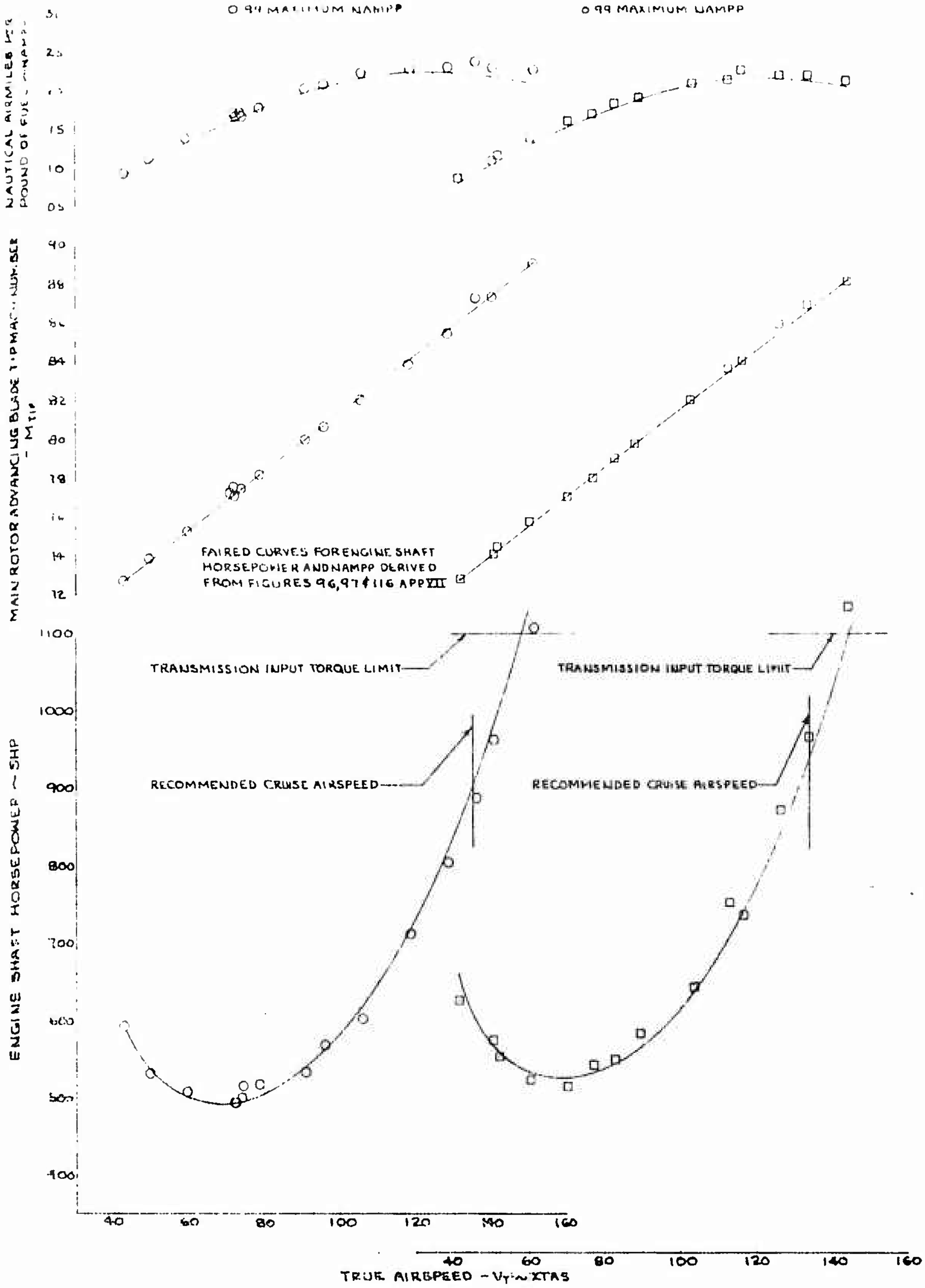


FIGURE NO 99

LEVEL FLIGHT PERFORMANCE

[illegible] G USA [illegible]15247

T53 [illegible]

[illegible]	AVG ALTITUDE ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT [illegible]
[illegible]	[illegible]	8090	200 (AFT)	0.005382	323	[illegible]
[illegible]	[illegible]00	8510	199.5 (AFT)	0.005762	321.3	[illegible]

NOTE: ALL ROCKET POD FAIRINGS REMOVED

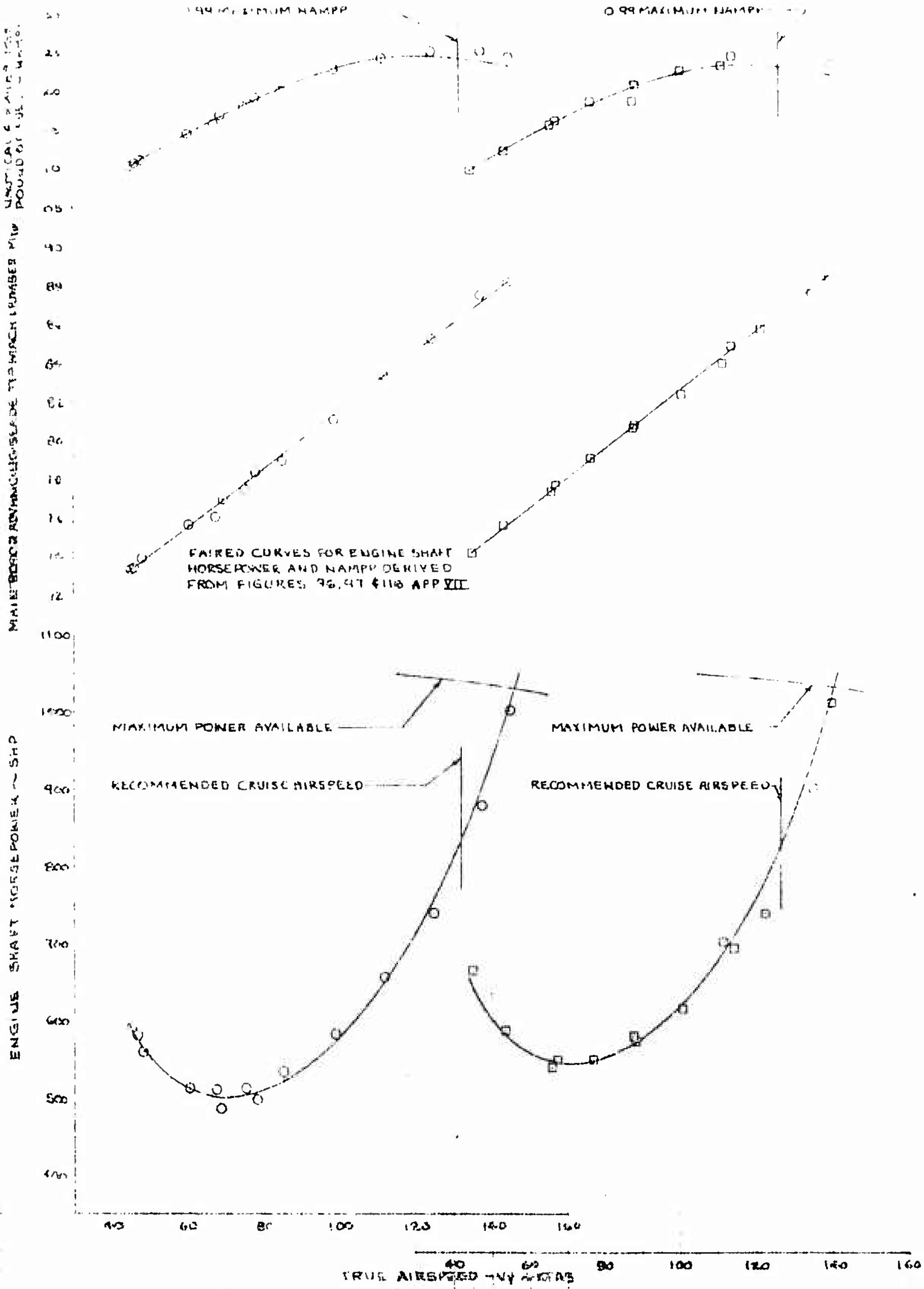


FIGURE NO 100
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13B/LE14001

SYM	AVG ALTITUDE H_D ~ FEET	AVG GROSS WEIGHT ~ LB	AVG LONG CG ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	9650	9020	199.7 (AFT)	0.006036	323	HVY HOG
□	14550	8100	200.8 (AFT)	0.006297	323	HVY HOG

NOTE: ALL ROCKET POD FAIRINGS REMOVED

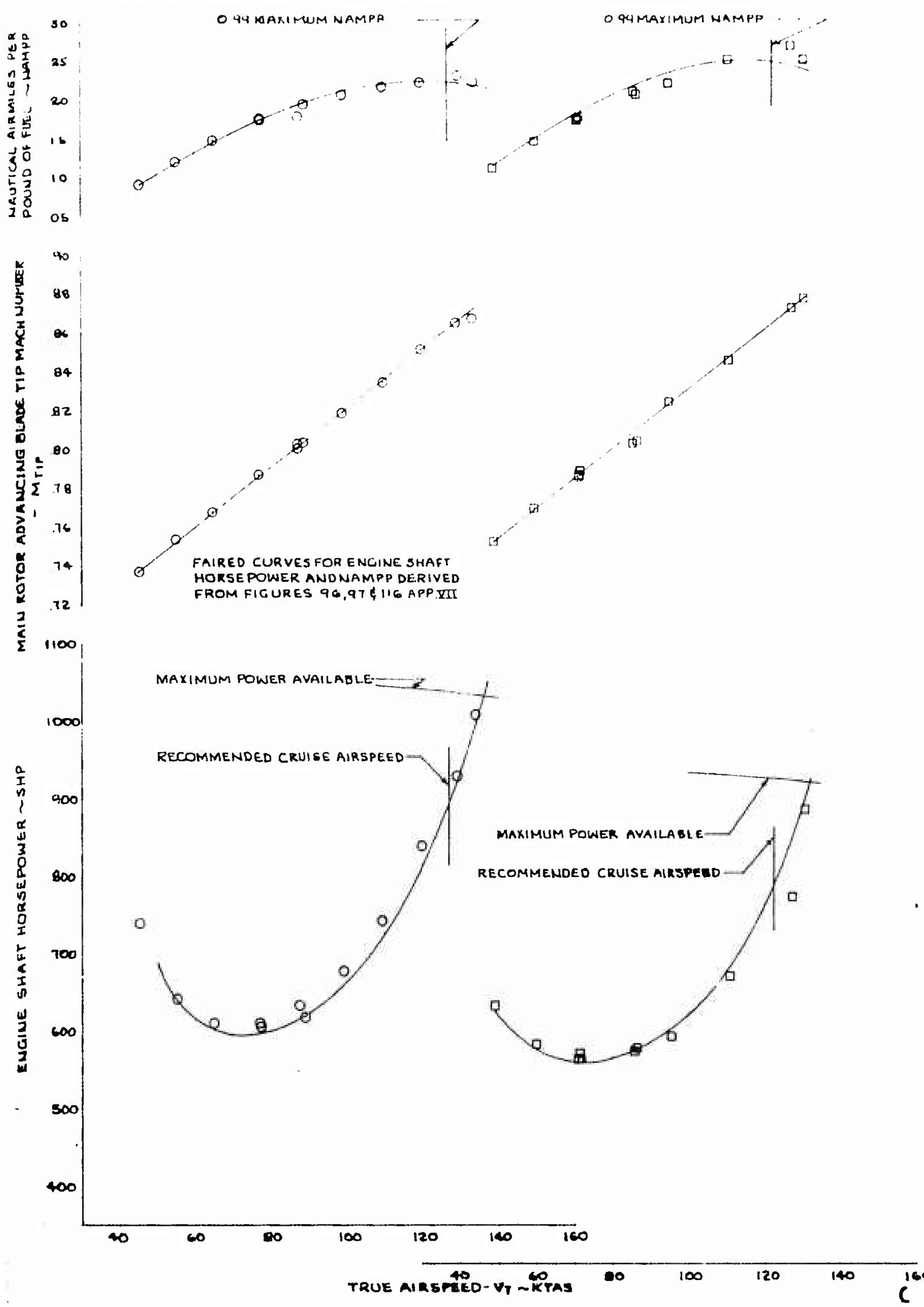


FIGURE NO 101

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYM	AVG ALTITUDE H_D ~ FT	AVG GROSS WEIGHT ~ LB	AVG LONG C.G. ~ IN	AVG THRUST COEFF ~ C_T	ROTOR SPEED ~ RPM	ARMAMENT CONFIG
○	14440	8610	199.9 (AFT)	0.006734	322.8	HVY HOG

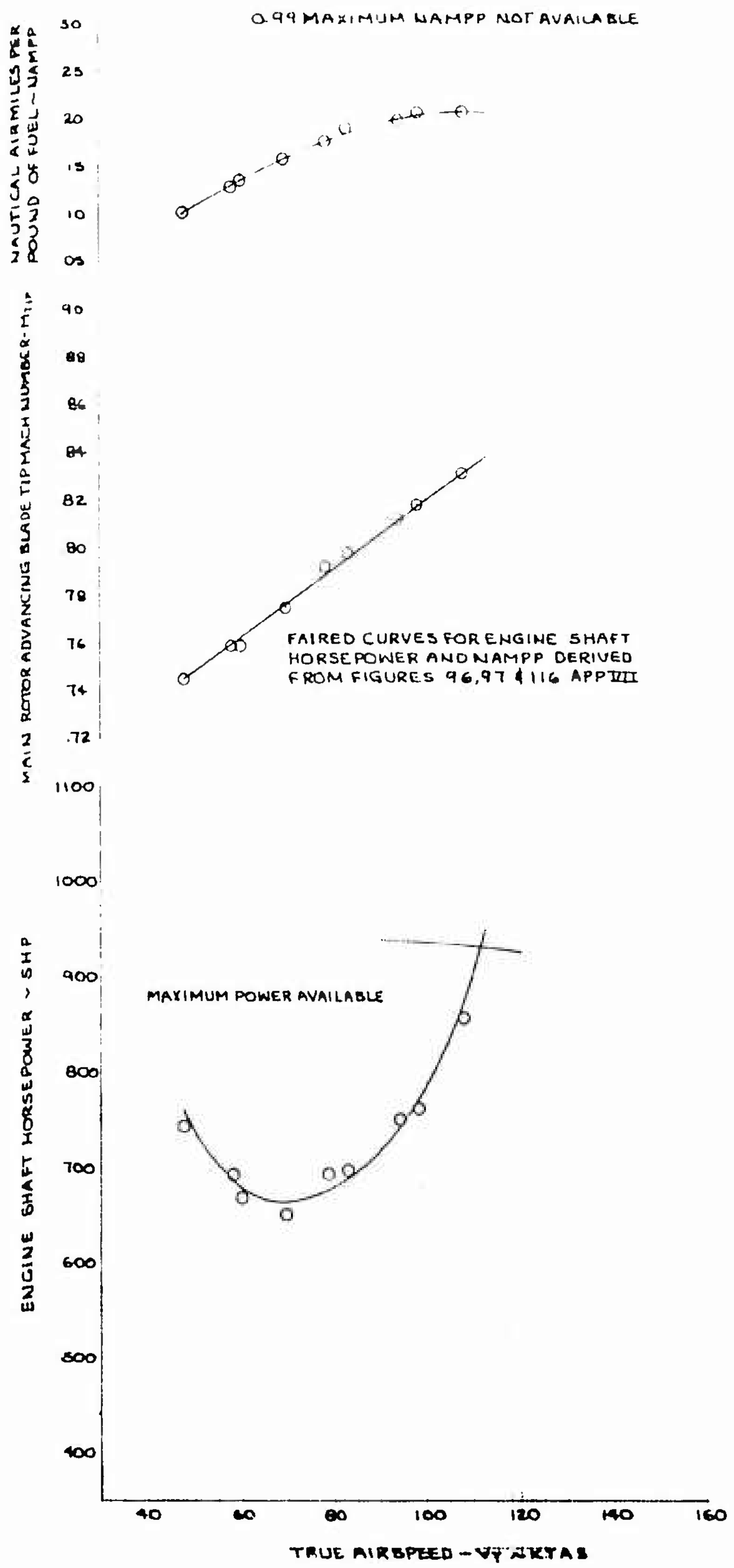


FIGURE No. 102

MAXIMUM AIRSPEED IN LEVEL FLIGHT

AH-1G T53-L-13

OUTB'D ALTERNATE WITH ROCKET POD FAIRINGS REMOVED

CONTRACT GUARANTEE COMPLIANCE CHECK

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = SEA LEVEL
GROSS WEIGHT = 8000 LB

NOTE: DATA DERIVED FROM FIGURES 38, 51, 52, 72, 73, 96, 97, & 118 APP VII

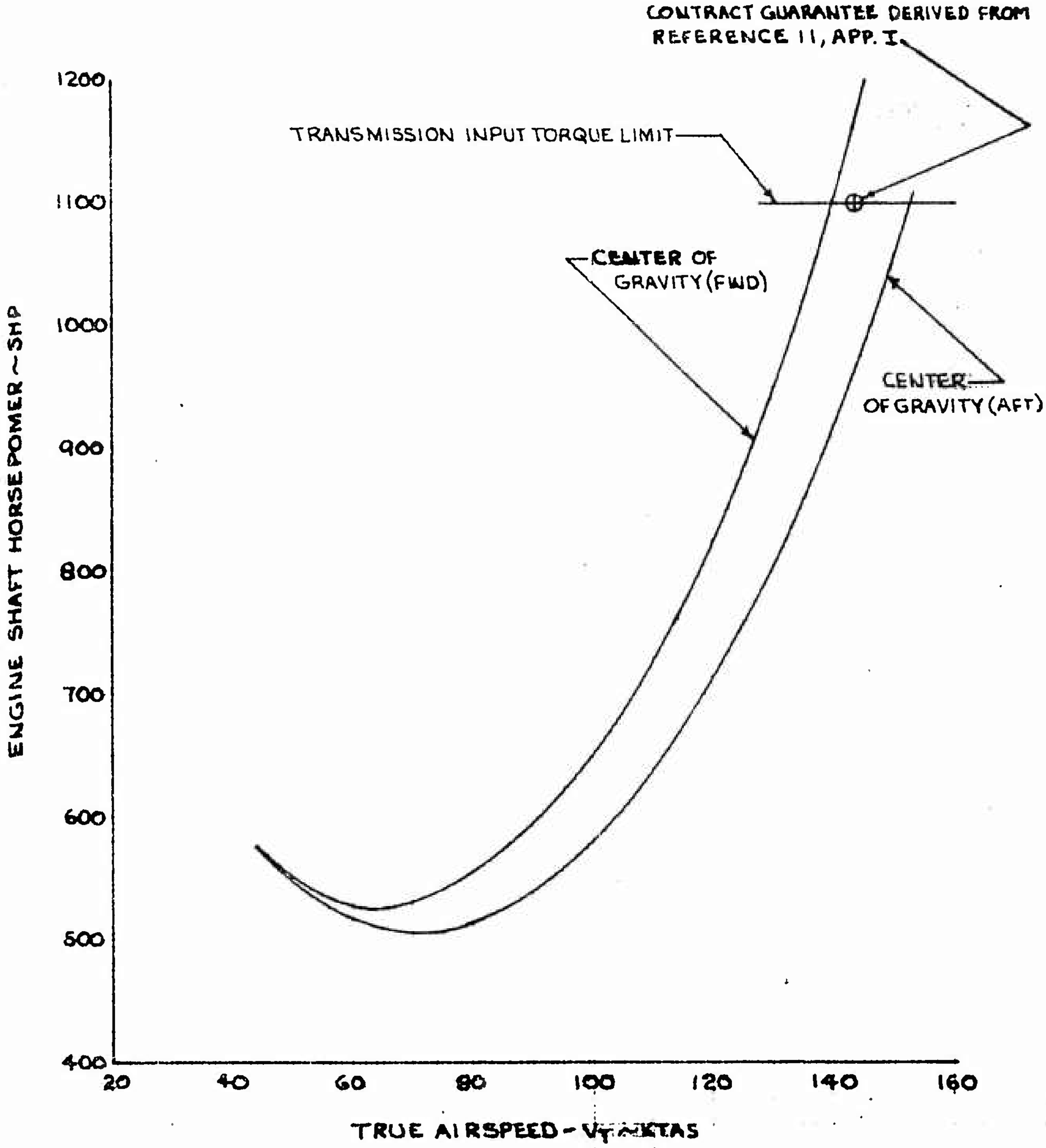


FIGURE NO 103
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
OUTBOARD ALTERNATE WITH ROCKET POD FAIRINGS REMOVED
CONTRACT GUARANTEE COMPLIANCE CHECK

INITIAL CONDITION	FINAL CONDITION
ROTOR SPEED = 324 RPM	ROTOR SPEED = [illegible]
DENSITY ALTITUDE = SEA LEVEL	DENSITY ALTITUDE = [illegible]
GROSS WEIGHT = 9175 LB	GROSS WEIGHT = [illegible]
CENTER OF GRAVITY = 191.9 INCHES (FWD)	CENTER OF GRAVITY = 190 [illegible] INCHES (FWD)

NOTE: THE CALCULATION OF NAMPP DOES NOT INCLUDE THE [illegible] CONSERVATIVE FUEL FLOW MARGIN

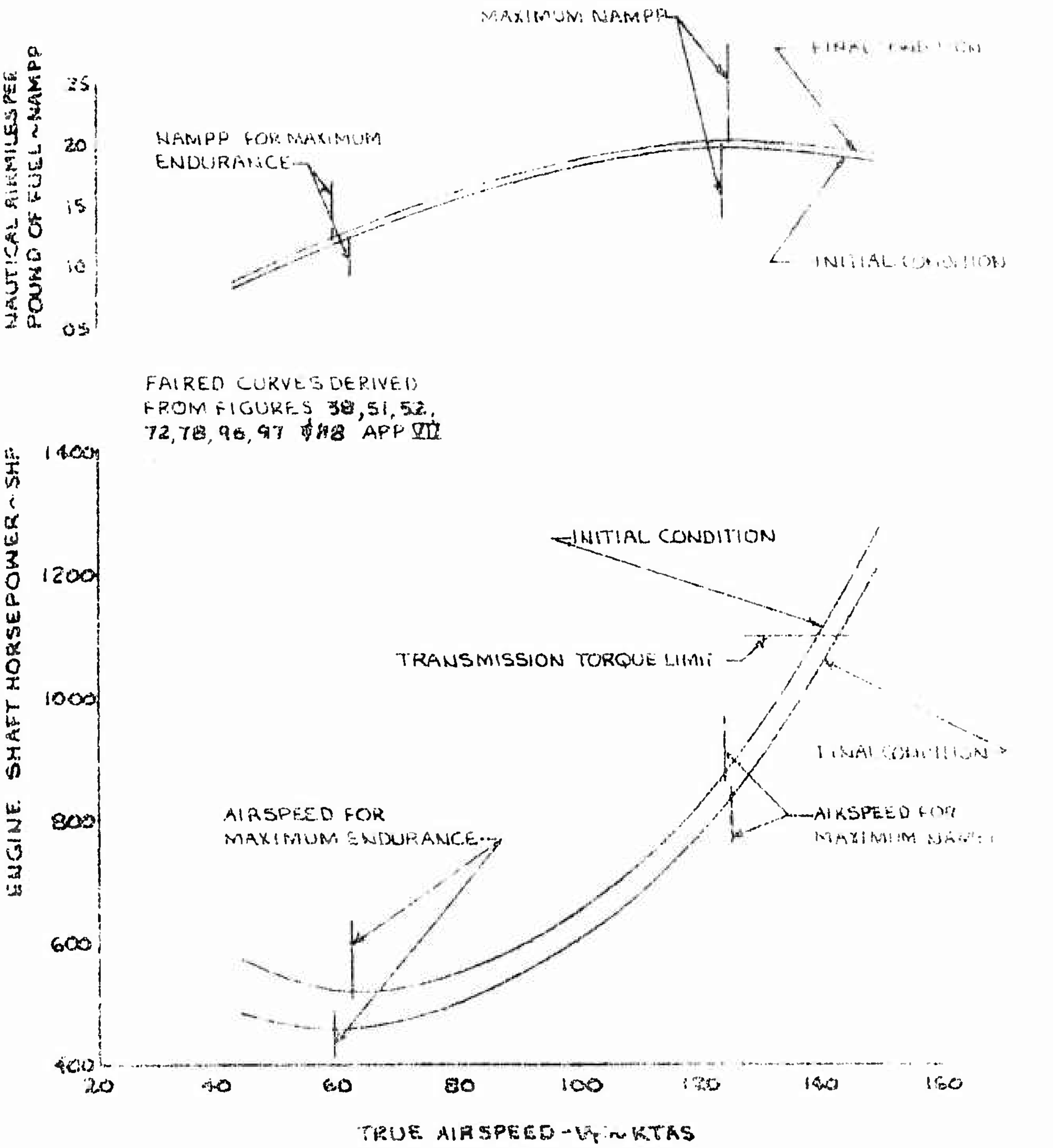
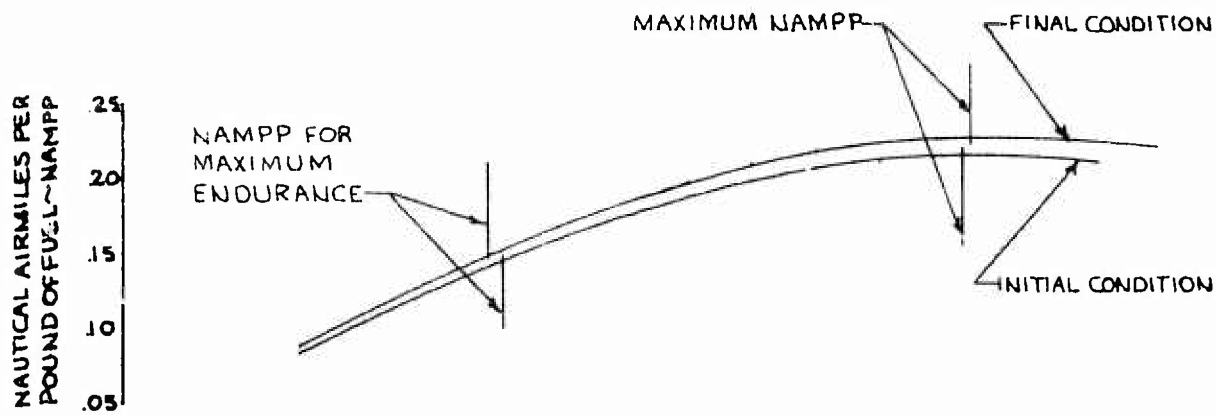


FIGURE NO. 104
LEVEL FLIGHT PERFORMANCE
AH-1G T53-L-13
OUTB'D ALTERNATE WITH ROCKET POD FAIRINGS REMOVED
CONTRACT GUARANTEE COMPLIANCE CHECK

INITIAL CONDITION:	FINAL CONDITION:
ROTOR SPEED = 324 RPM	ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = SEA LEVEL	DENSITY ALTITUDE = SEA LEVEL
GROSS WEIGHT = 7975 LB	GROSS WEIGHT = 6560 LB
CENTER OF GRAVITY = 201.0 INCHES (AFT)	CENTER OF GRAVITY = 201.0 INCHES (AFT)

NOTE: THE CALCULATION OF NAMPP DOES NOT INCLUDE THE 5 PERCENT CONSERVATIVE FUEL FLOW MARGIN



FAIRED CURVES DERIVED FROM FIGURES 38, 51, 52, 72, 73, 96, 97 & 118 APP. VIII

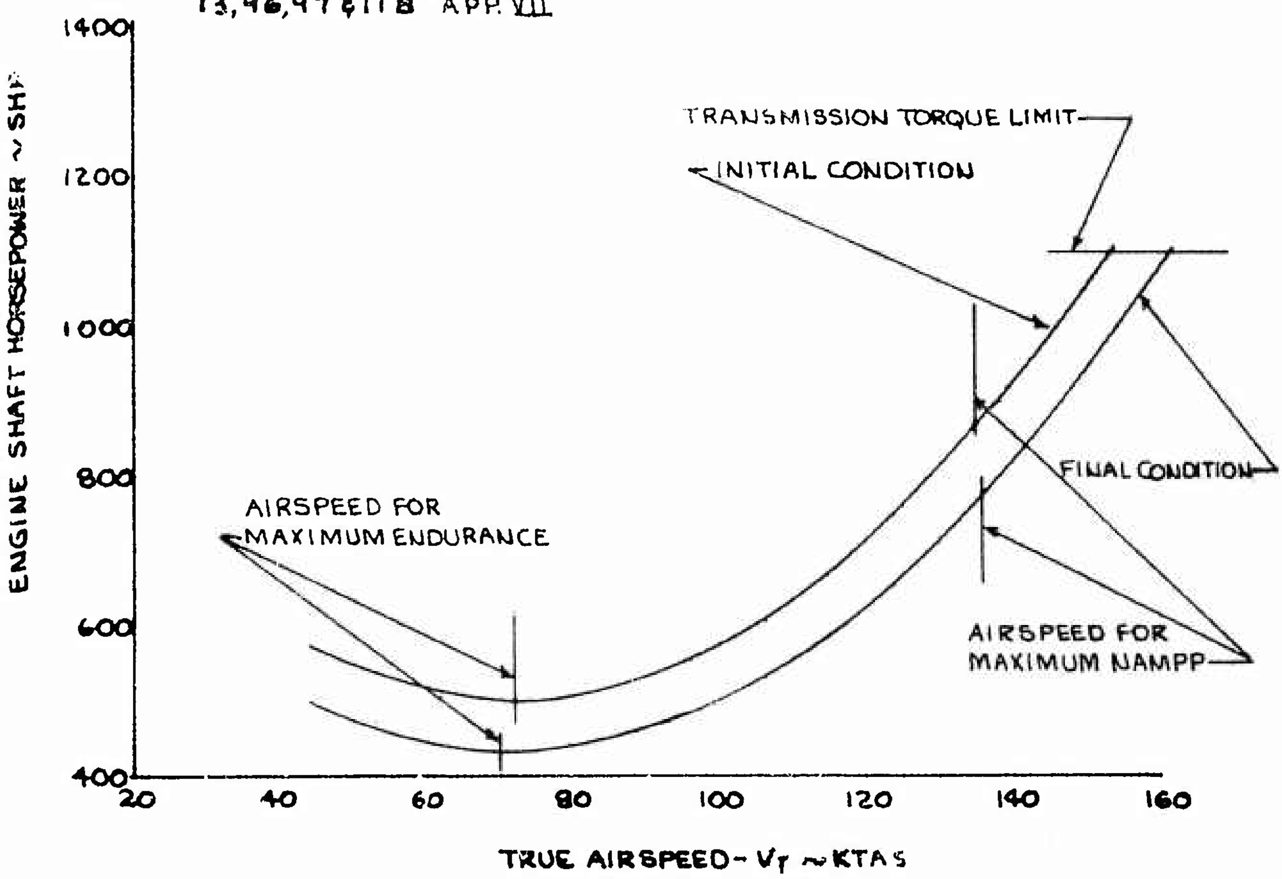


FIGURE NO. 105

COMPRESSIBILITY EFFECTS ON LEVEL FLIGHT PERFORMANCE

AH-1G T53-L-13
CLEAN CONFIGURATION
CENTER OF GRAVITY - FWD
AVG. THRUST COEFFICIENT = 0.006035
ROTOR SPEED = 324 RPM

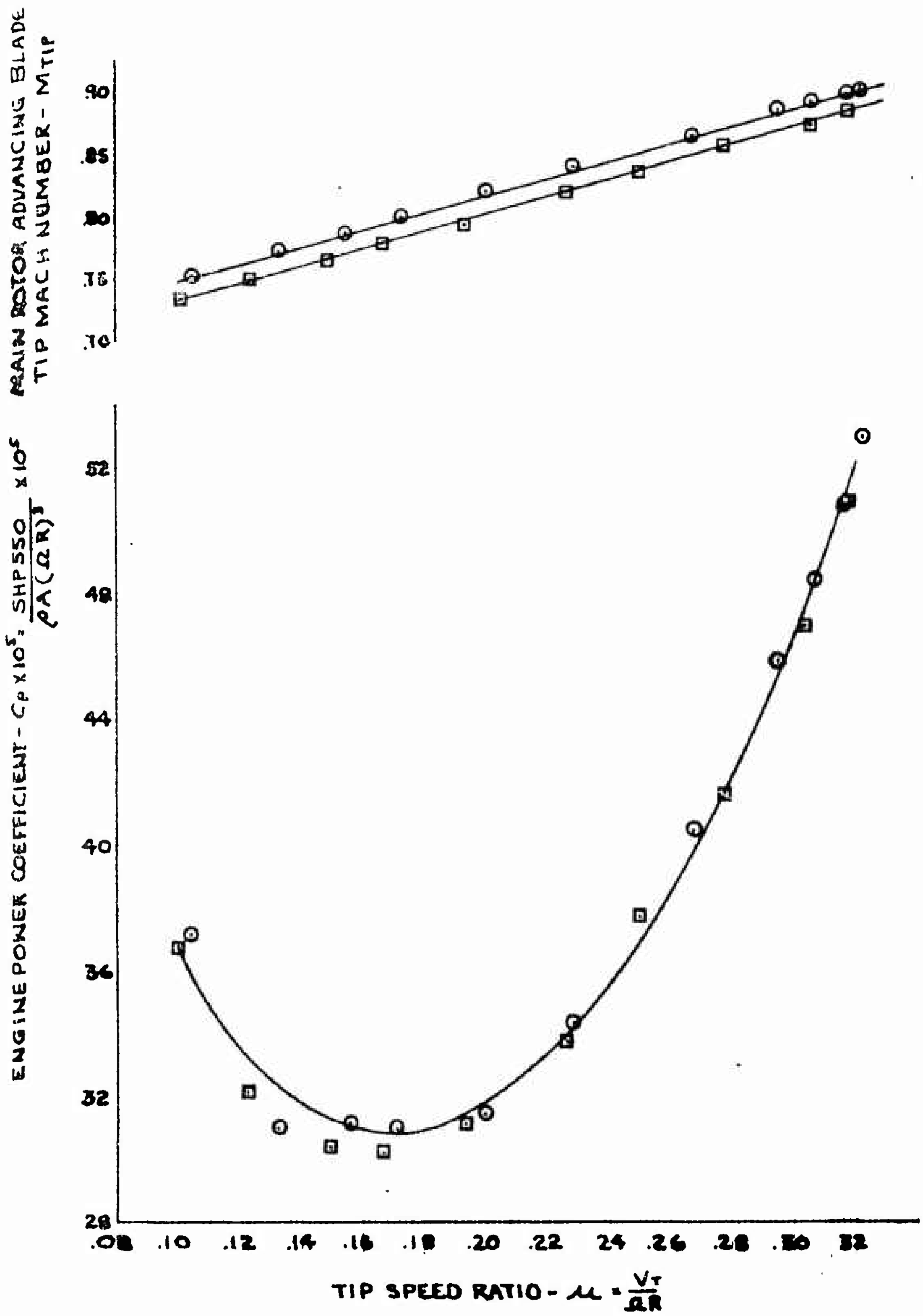


FIGURE NO 106

SUMMARY OF ENDURANCE, SPECIFIC RANGE, AND MAXIMUM AIRSPEED FOR LEVEL FLIGHT

AH-1G T53-L-13

CLEAN CONFIGURATION

CENTER OF GRAVITY - FORWARD

STANDARD DAY

ROTOR SPEED - 324 RPM

LEGEND	ALTITUDE ~ FT
————	SEA LEVEL
— — —	5000
—·—·—	10000

NOTE: CURVE DERIVED FROM FIGURES 39, 40 & 116, APP VII

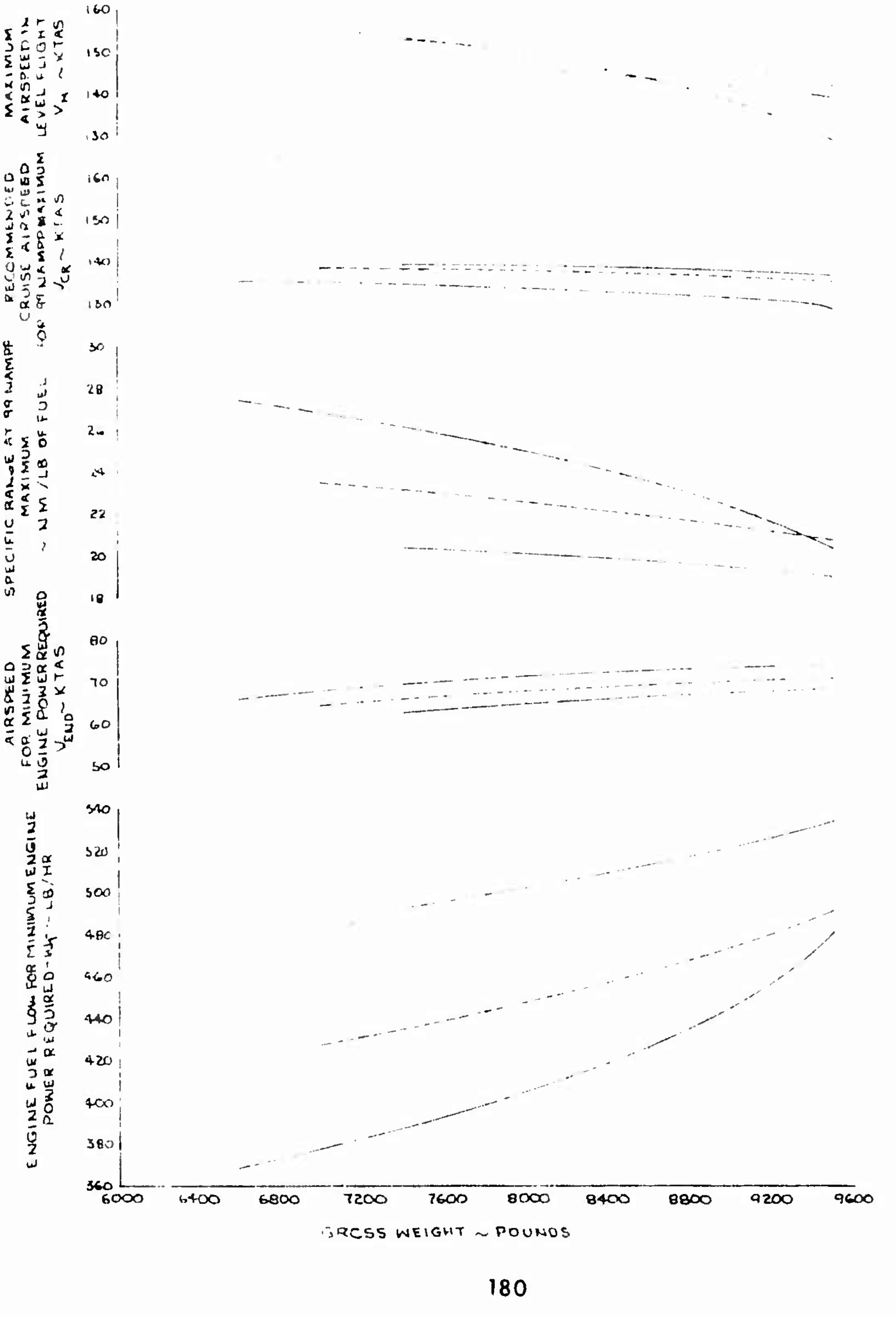


FIGURE NO. 101

SUMMARY OF ENDURANCE SPECIFIC RANGE AND MAXIMUM AIRSPEED FOR LEVEL FLIGHT

AH-1G T53-L-13
CLEAN CONFIGURATION
CENTER OF GRAVITY: AFT
STANDARD DAY
ROTOR SPEED = 324 RPM

LEGEND ALTITUDE FT
SEA LEVEL
5000
10000

NOTE: CURVE DERIVED FROM FIGURES 51, 82 & 116 APP VII

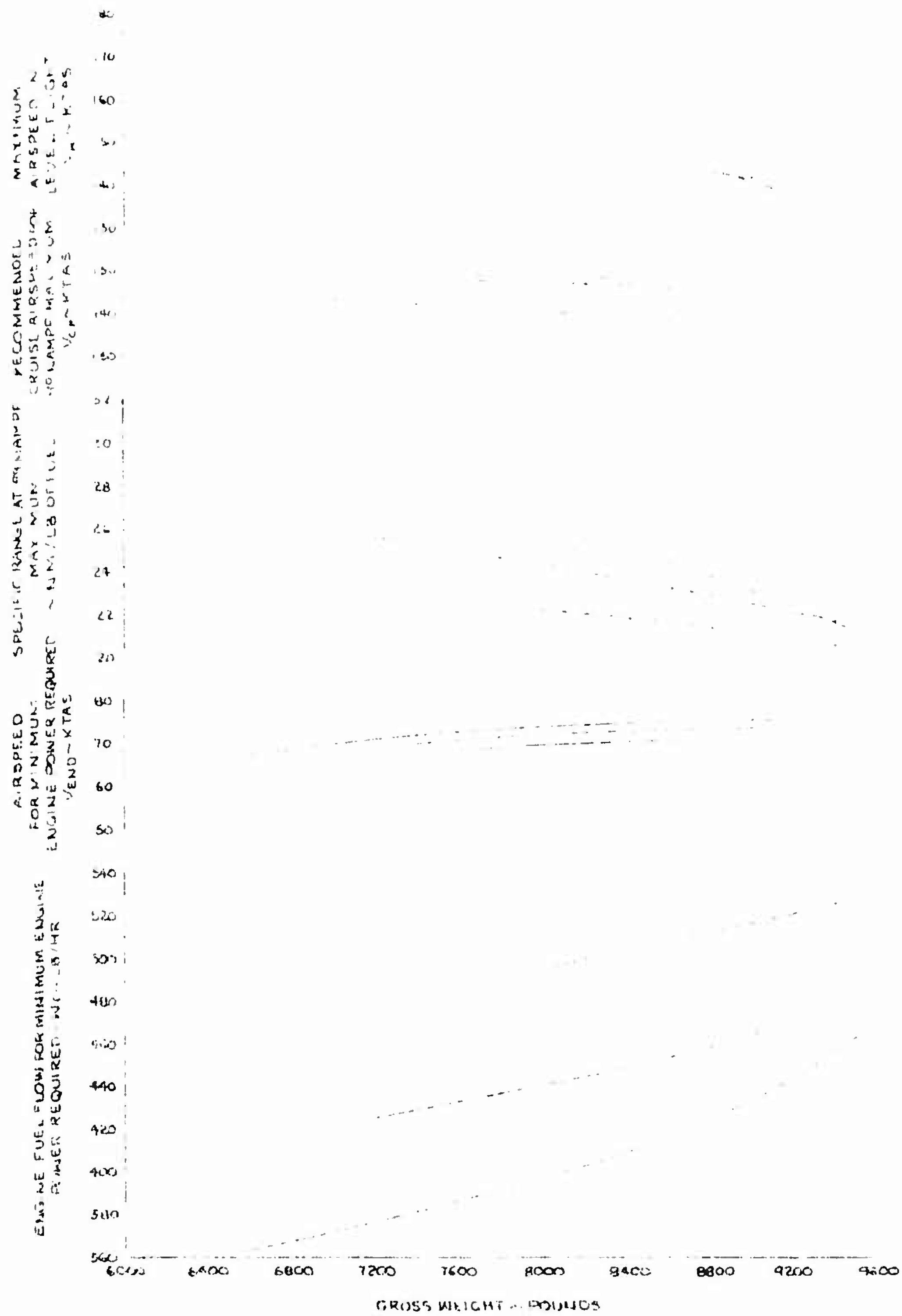


FIGURE No 108

SUMMARY OF ENDURANCE, SPECIFIC RANGE, AND MAXIMUM AIRSPEED FOR LEVEL FLIGHT

AH-1G T53-L-13

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY FORWARD

STANDARD DAY

ROTOR SPEED = 324 RPM

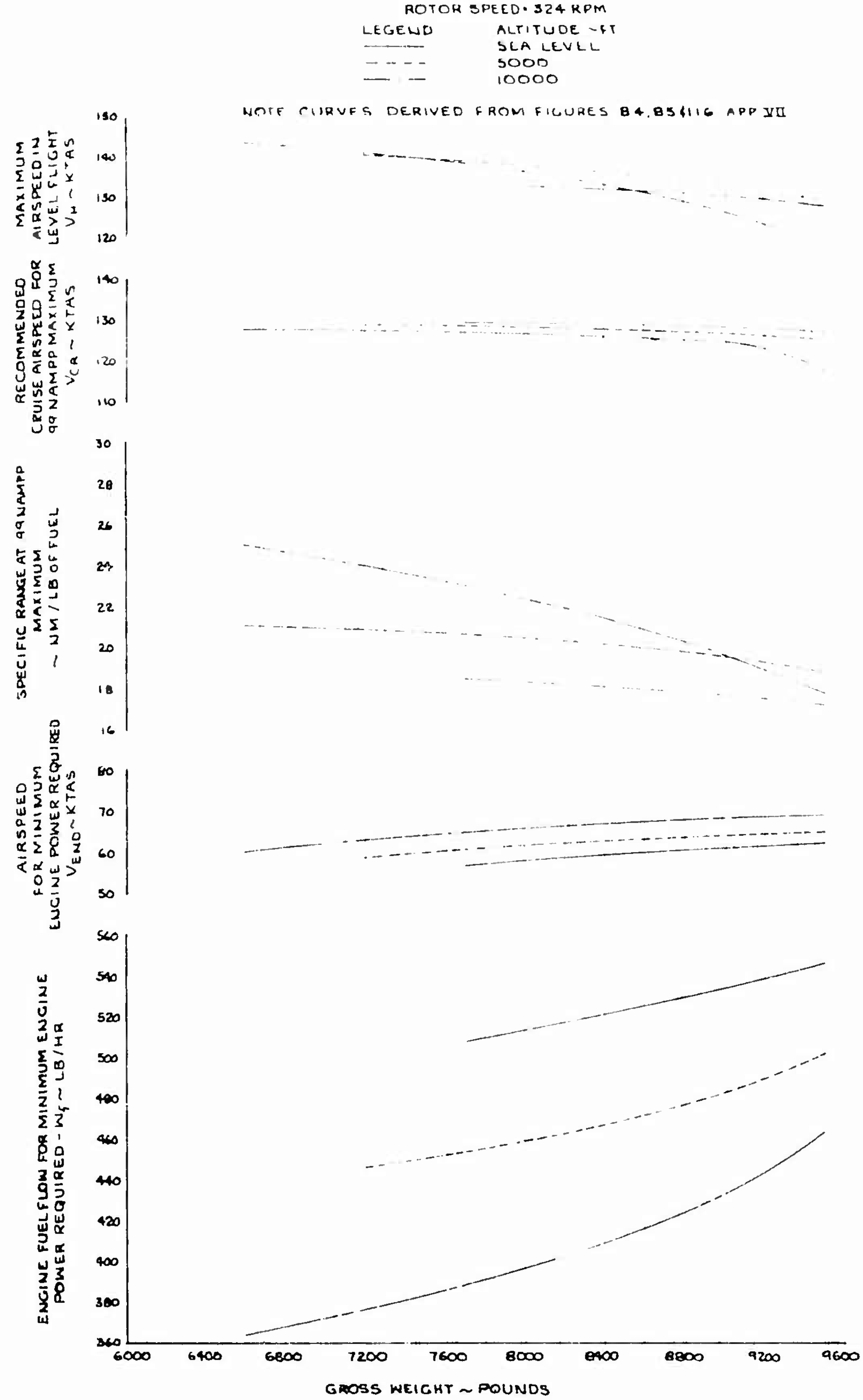


FIGURE No 109

SUMMARY OF ENDURANCE, SPECIFIC RANGE, AND MAXIMUM AIRSPEED FOR LEVEL FLIGHT

AH-1G T53-L-13

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

CENTER OF GRAVITY: AFT

STANDARD DAY

ROTOR SPEED: 324 RPM

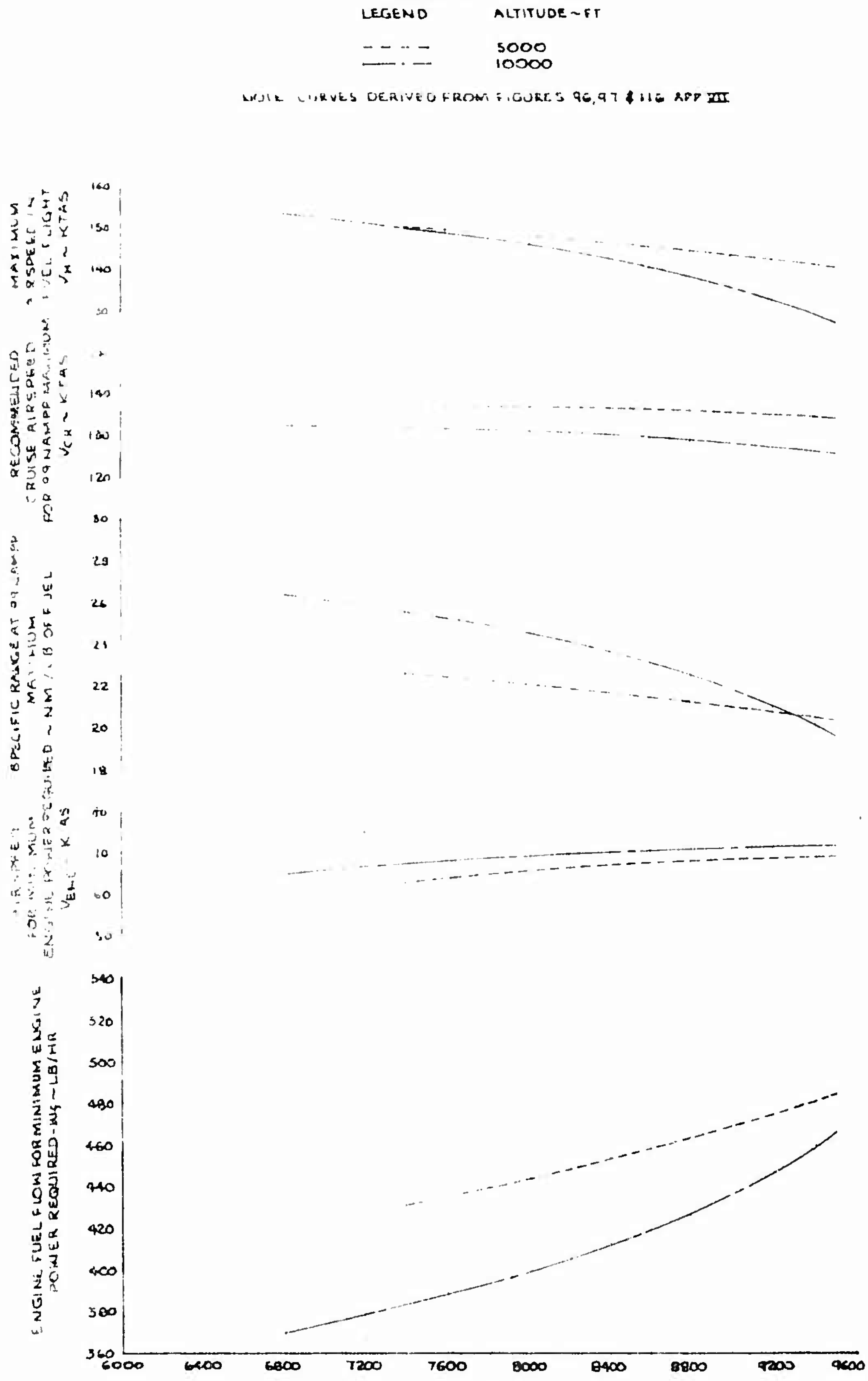


FIGURE No. 110
AUTOROTATIONAL DESCENTS
AH-1G USA S/N 615247
CLEAN CONFIGURATION

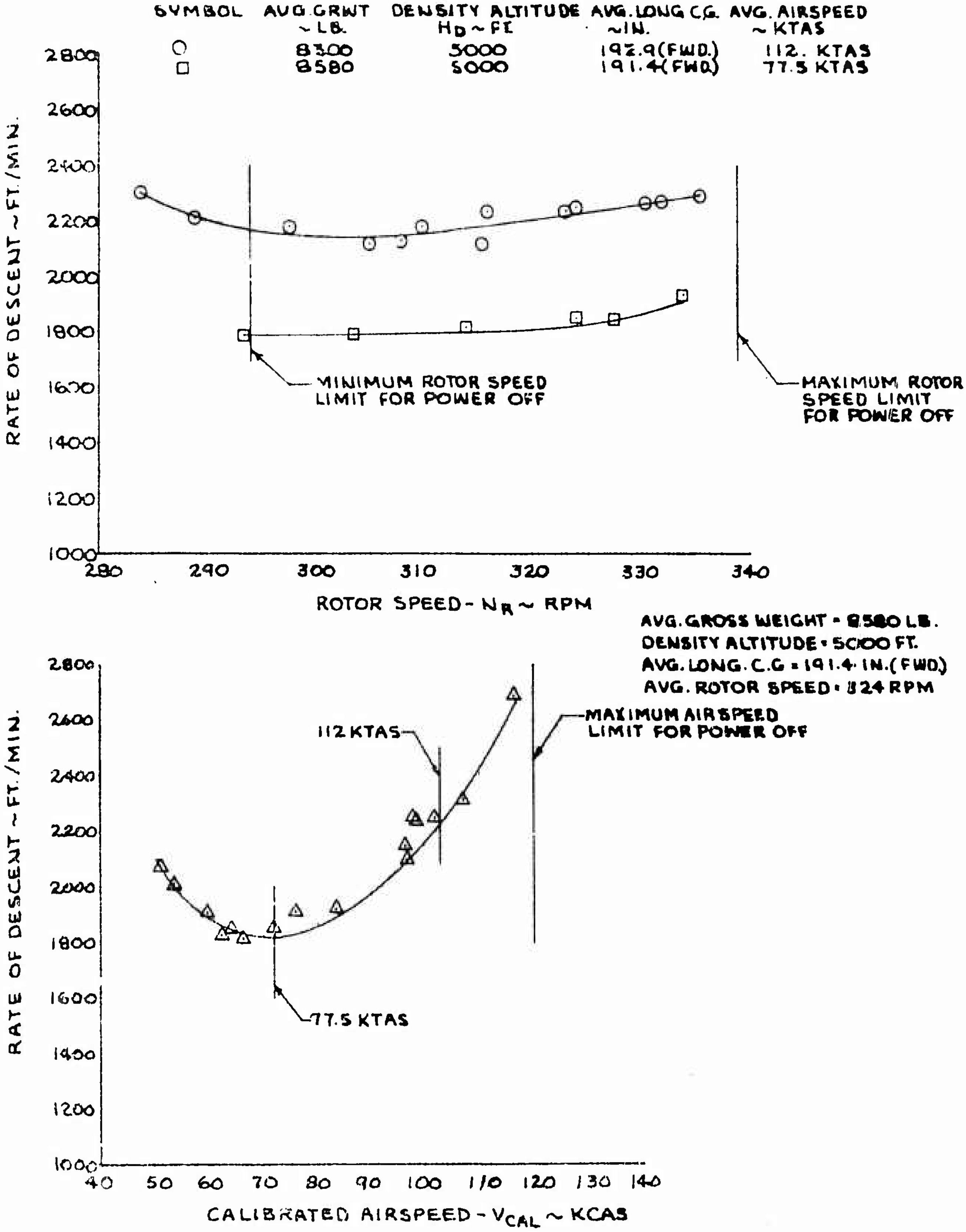
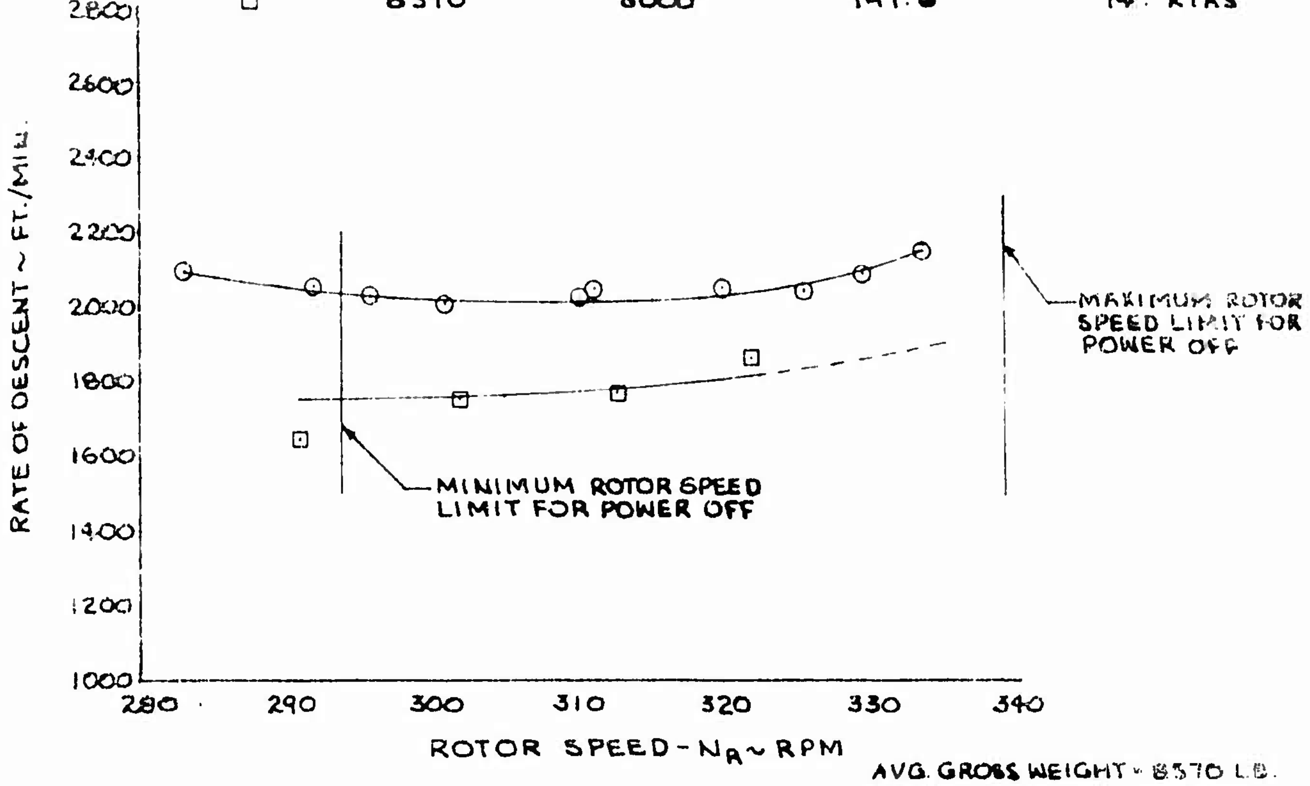


Figure No. 111
Autorotational Descents
AH-1G USA S/N 615247
HVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYMBOL	AVG GRWT ~LB	DENSITY ALTITUDE H_D ~ FT.	AVG. LONG. C.G. ~IN.	AVG. AIRSPEED ~KTAS
○	8400	5000	192.5	98. KTAS
□	8570	5000	191.6	74. KTAS



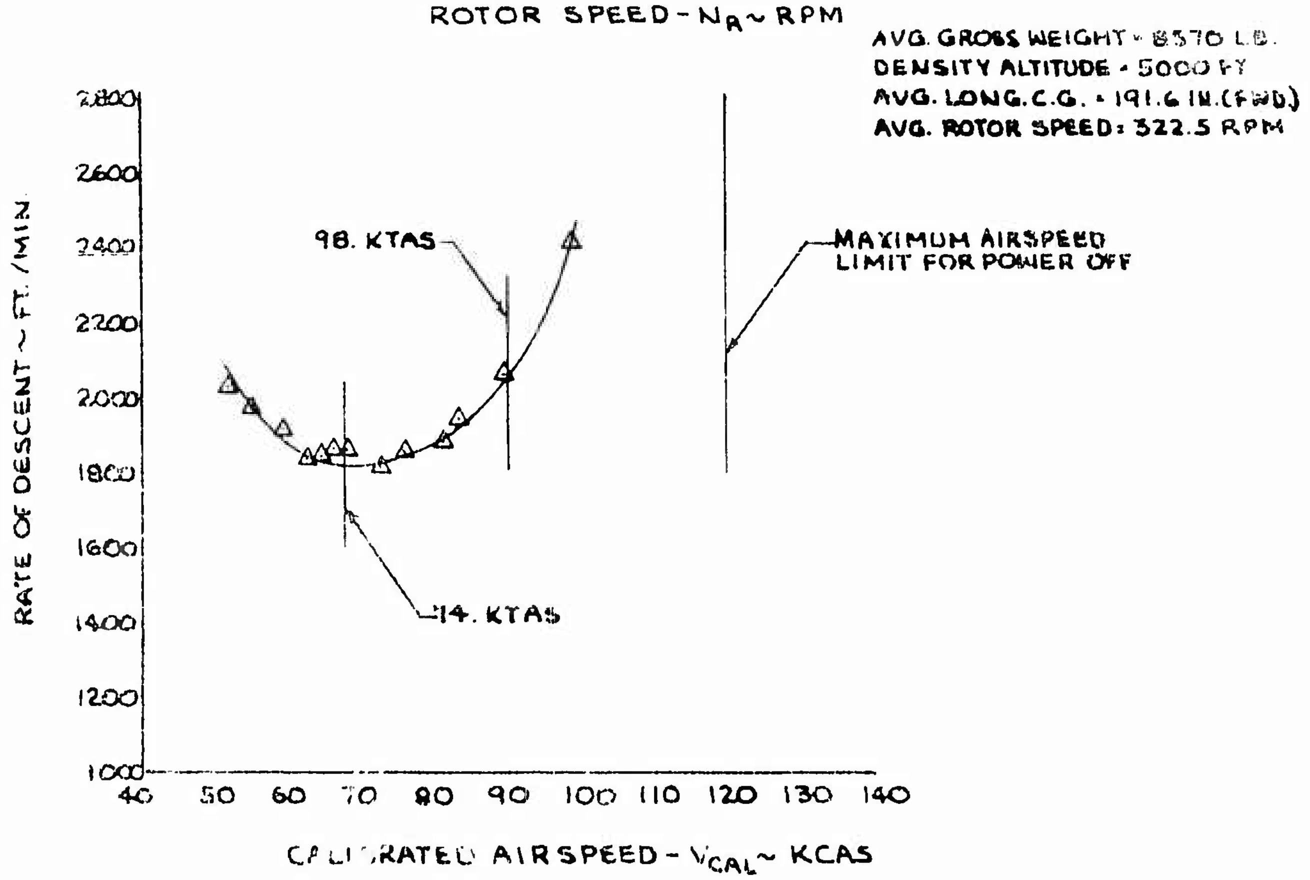


FIGURE NO. 112

LANDING PERFORMANCE

AH-1G USA #615247

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

ROTOR SPEED = 324 RPM

DENSITY ALTITUDE - 6360 FT.

SYM	GR WT ~ LB.	LONG. C.G. ~ IN.
○	8490	195.7
△	8410	195.6
□	9240	195.4
◇	9500	195.2

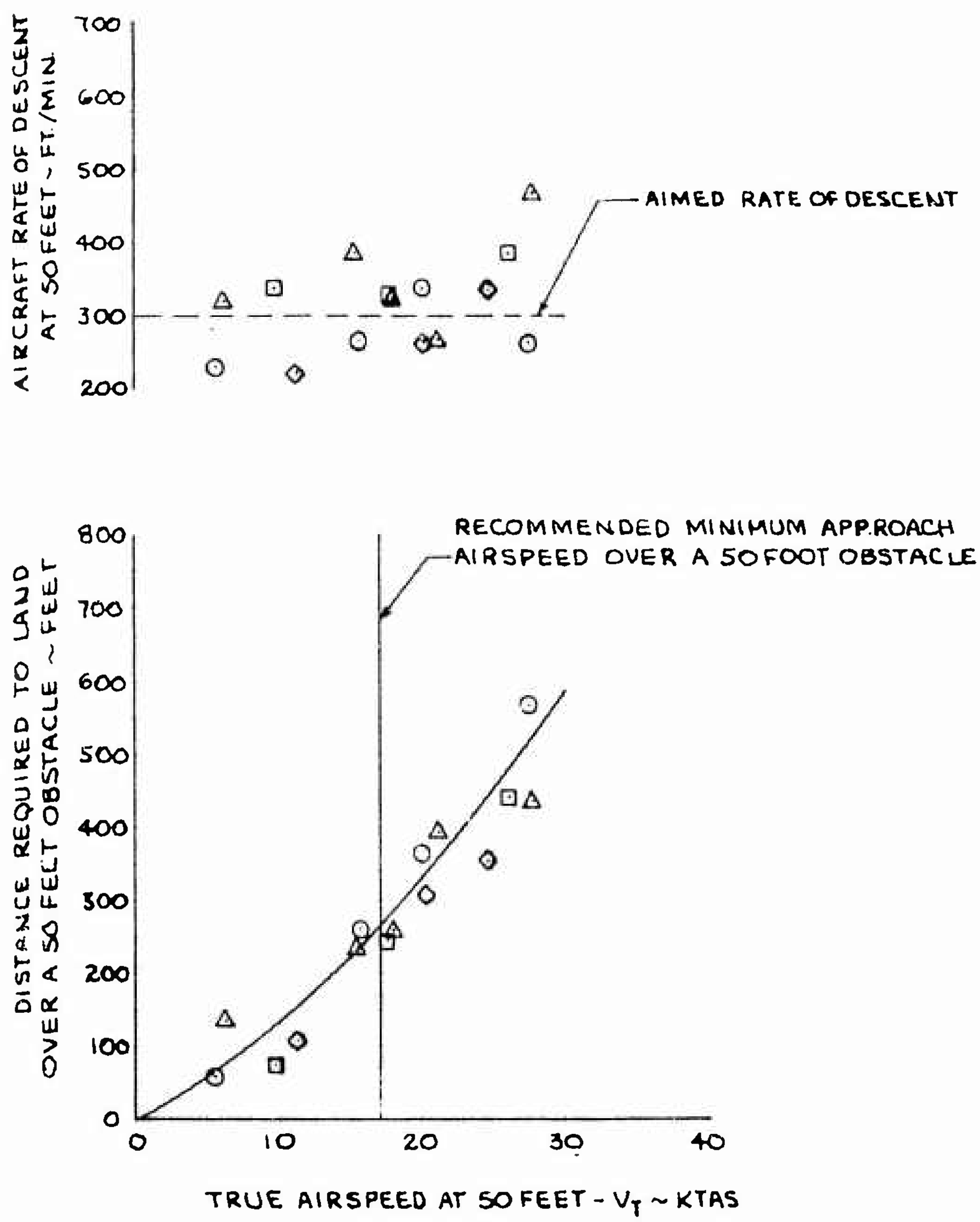


FIGURE No. 113
ENGINE INLET CHARACTERISTICS
AH-1G T53-L-13 ENGINE
ENGINE PARTICLE SEPARATOR INSTALLED

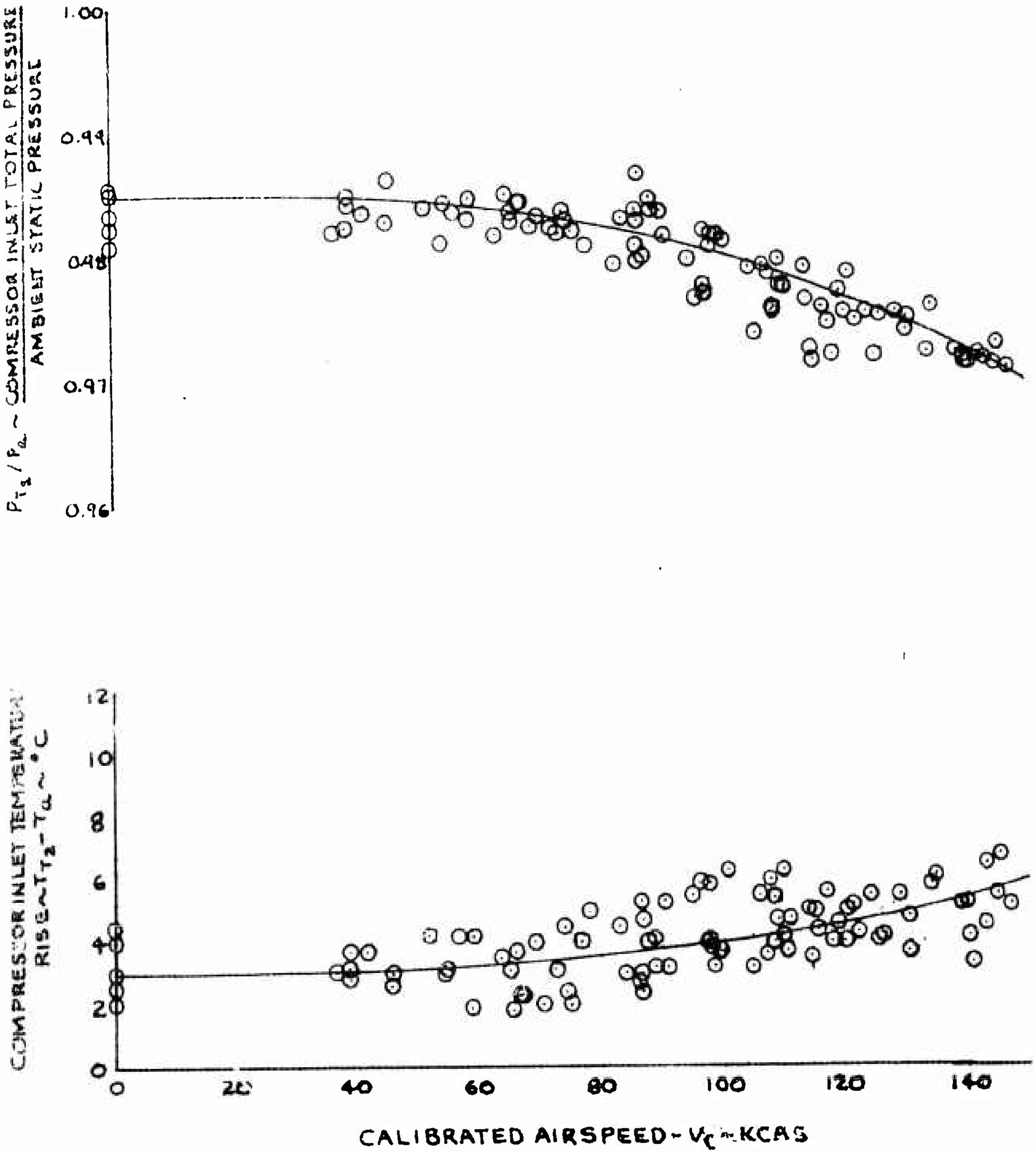


Figure No. 114

Military Limit Shaft Horsepower Available

AH-1G T53-L-13

HOVERING

NOTES: 1 ENGINE PARTICLE SEPARATOR INSTALLED
2 DATA BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 104.33
3 ENGINE INLET CHARACTERISTICS BASED ON FIGURE 113 APP. VII
4 GENERATOR ELECTRICAL LOAD = ZERO
5. PERCENT AIRBLEED (W_{bl}/W_a) = 0.6%
6. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR
7 ROTOR SPEED = 324 RPM

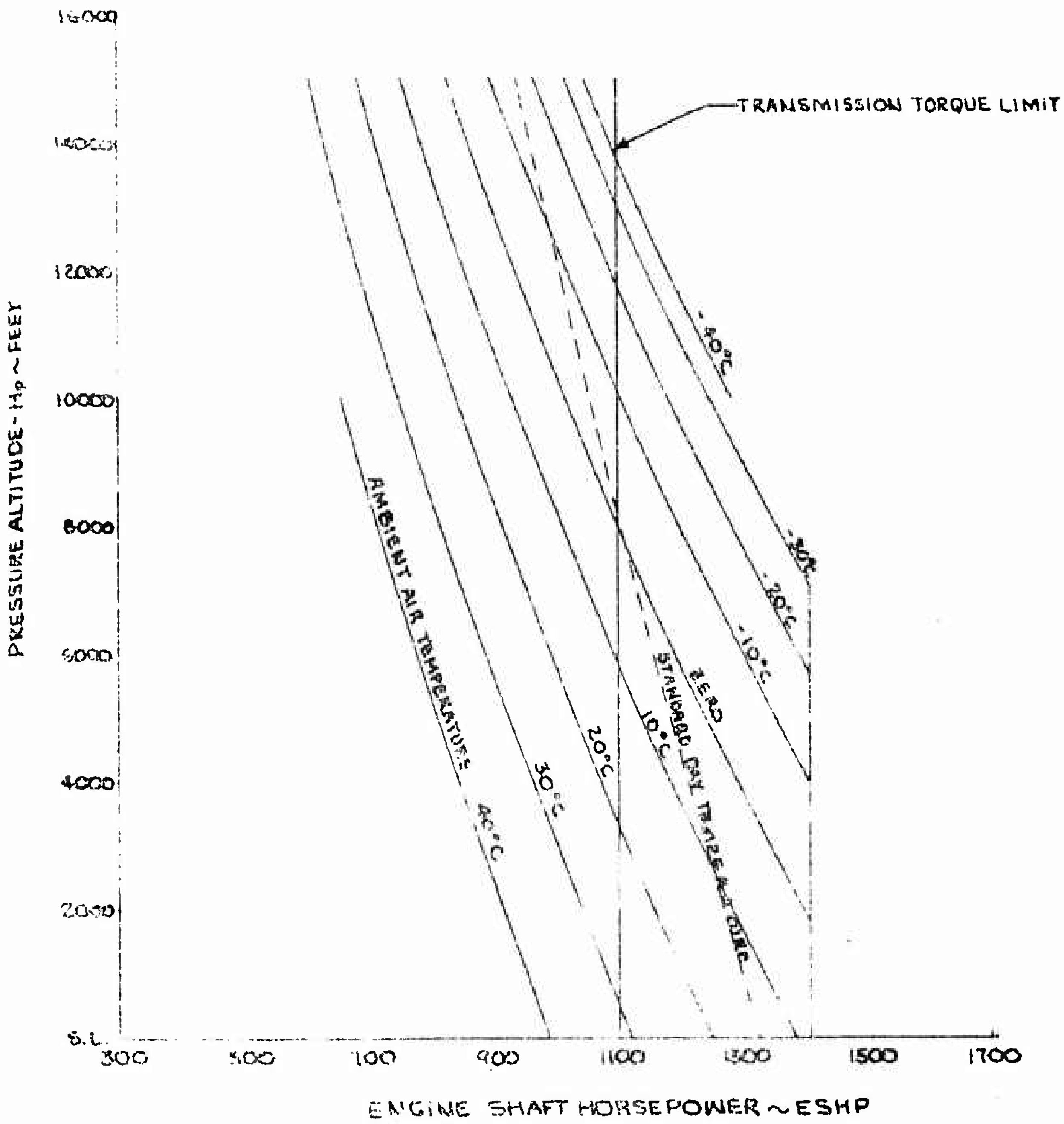


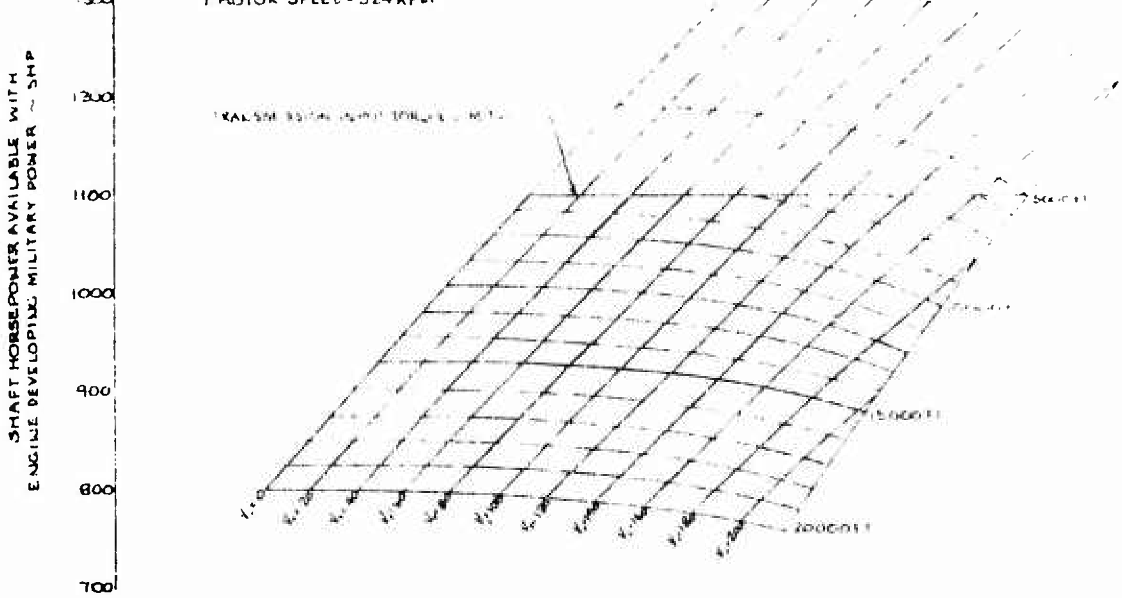
FIGURE NO 115

SPECIFICATION SHAFT HORSEPOWER AVAILABLE AND FUEL FLOW

AH 1G T53-L-13

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES 1 CURVES BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 104 53
2 ENGINE INLET CHARACTERISTICS BASED ON FIGURE NO 113 APP VII
3 GENERATOR ELECTRICAL LOAD = ZERO
4 PERCENT AIRBLEED(N_{bl}/W_a) = ZERO
5 ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING
6 ICAO STANDARD DAY
7 ROTOR SPEED = 324 RPM



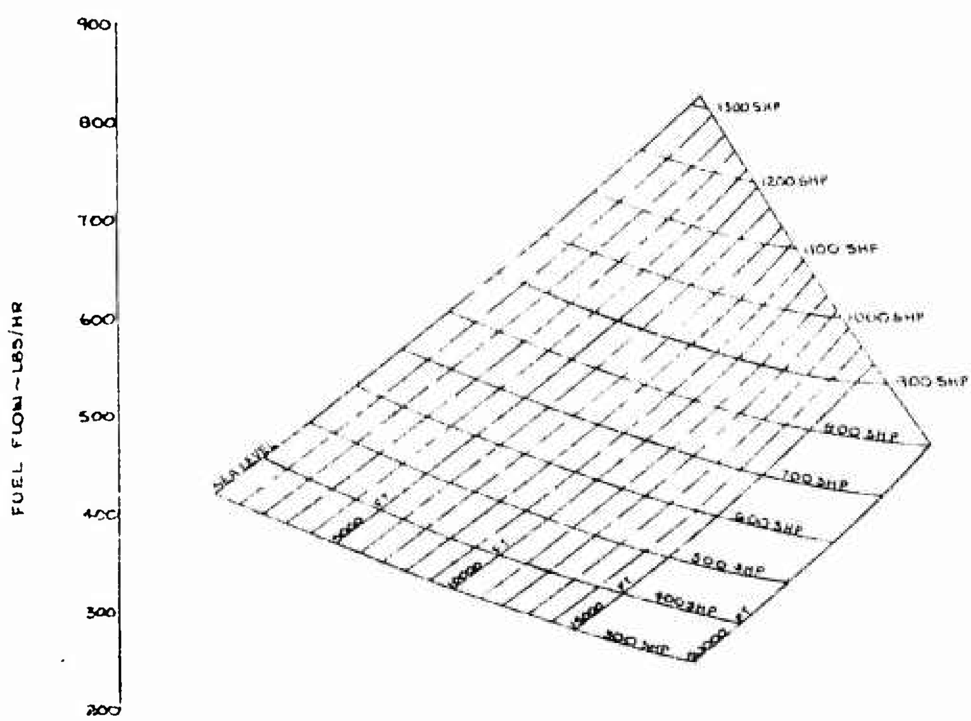


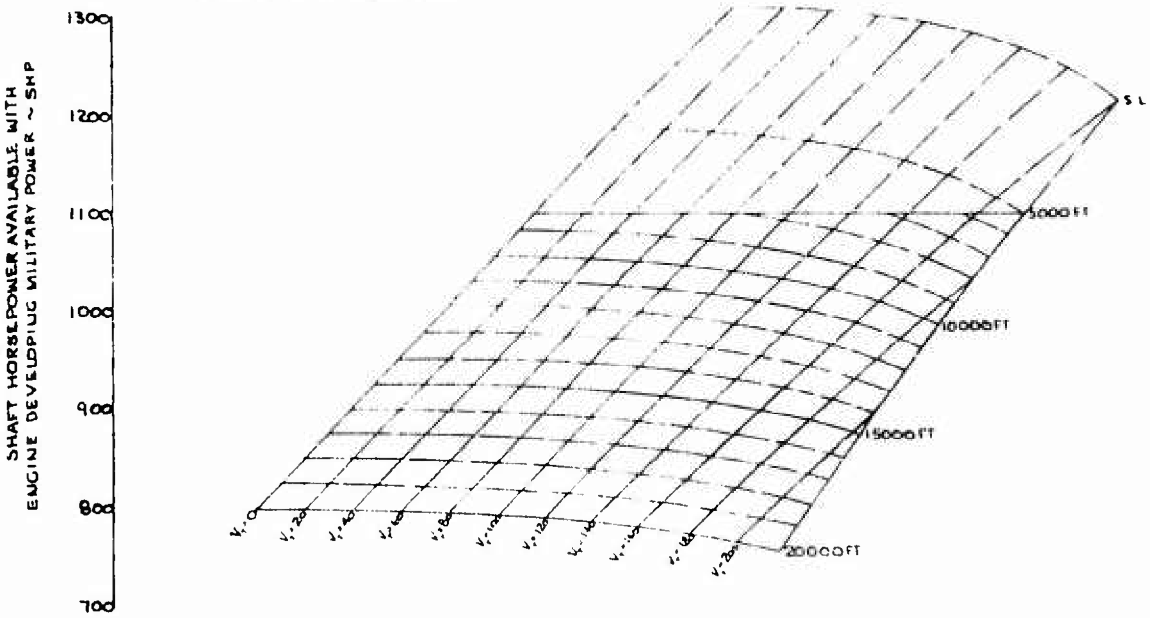
FIGURE No. 116

SPECIFICATION SHAFT HORSEPOWER AVAILABLE AND FUEL FLOW

AH-1G T53-L-13

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1 CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 140.53
2 ENGINE INLET CHARACTERISTICS BASED ON FIGURE NUMBER 113 APP VII
3 GENERATOR ELECTRICAL LOAD = ZERO
4 PERCENT AIRBLEED (W_{bl}/W_a) = 0.67
5 ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR
6 COCKPIT AIR CONDITIONING DRIVEN BY ENGINE BLEED AIR
7 ICAO STANDARD DAY
8 ROTOR SPEED = 324 RPM



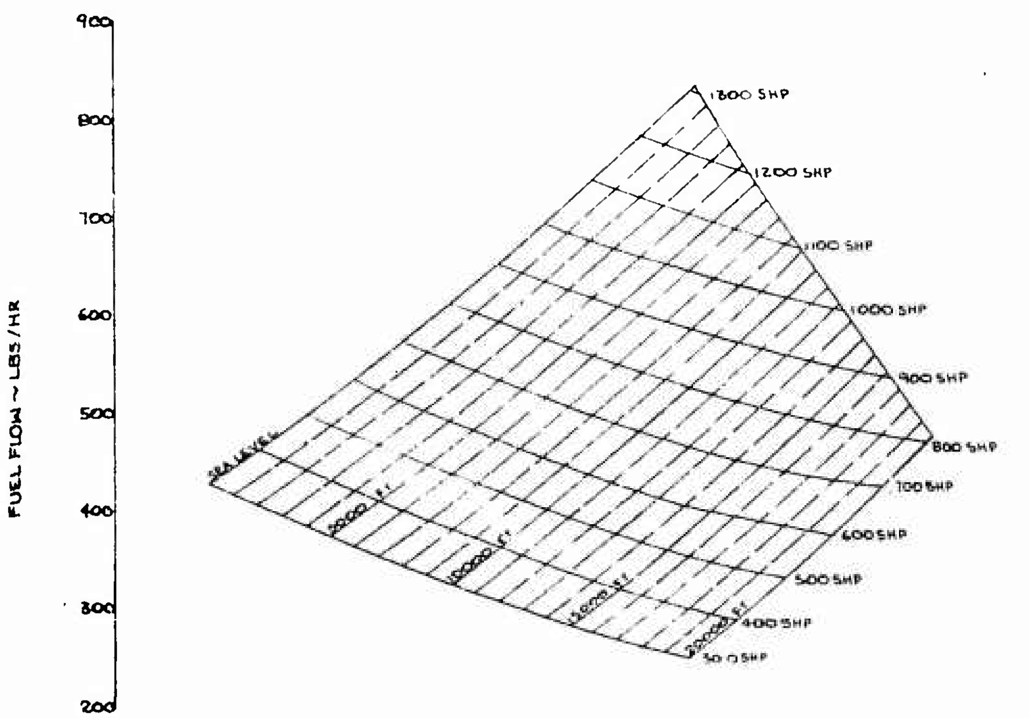


FIGURE NO. 117

SPECIFICATION SHAFT HORSEPOWER AVAILABLE AND FUEL FLOW

AH-1G T53-L-13

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 104 53

2. ENGINE INLET CHARACTERISTICS BASED ON FIGURE NO. 113 APP VII

3. GENERATOR ELECTRICAL LOAD = ZERO

4. PERCENT AIRBLEED (W_{a_1}/W_{a_2}) = 3.6%

5. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR

6. ICAO STANDARD DAY

7. ROTOR SPEED = 324 RPM

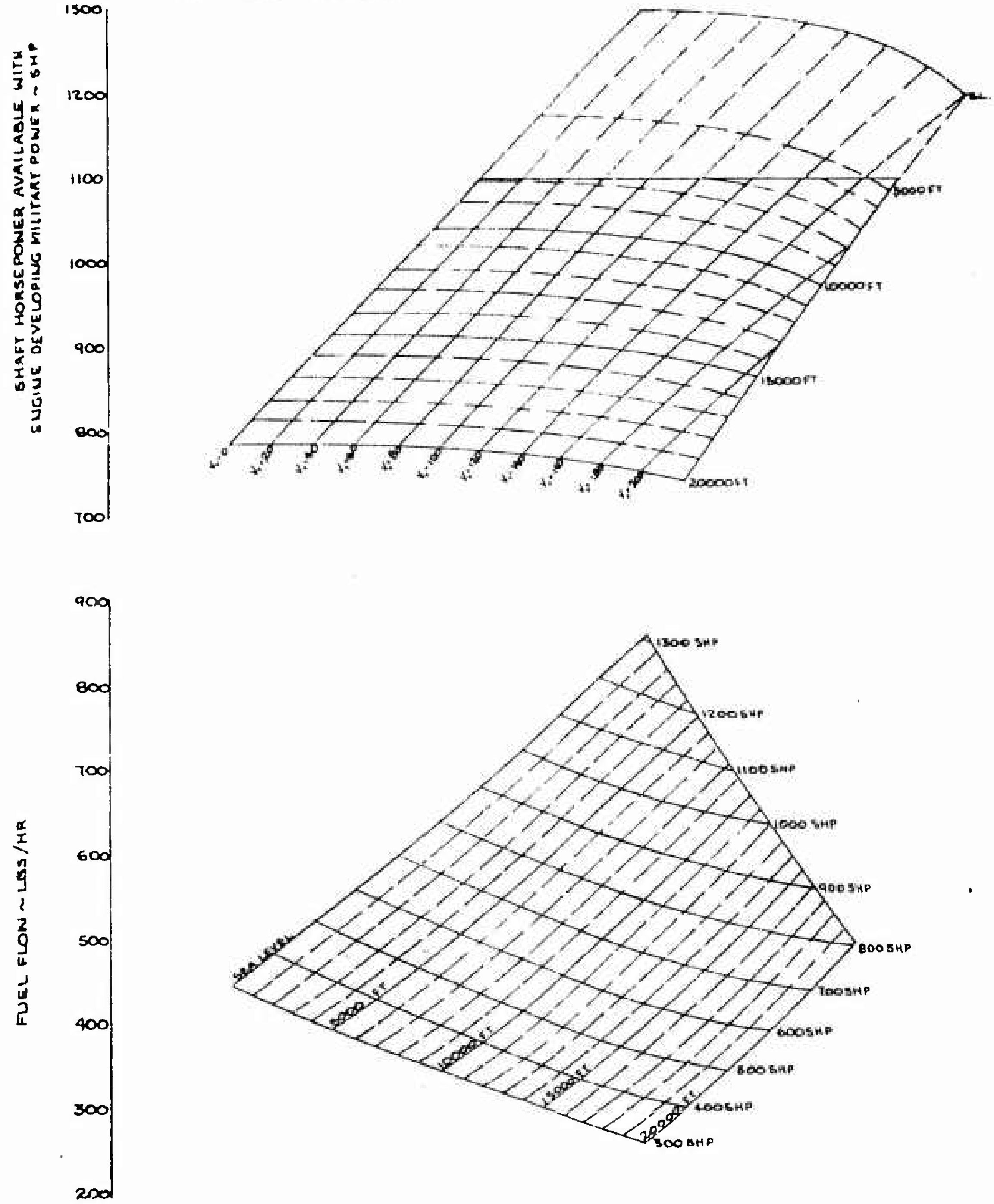
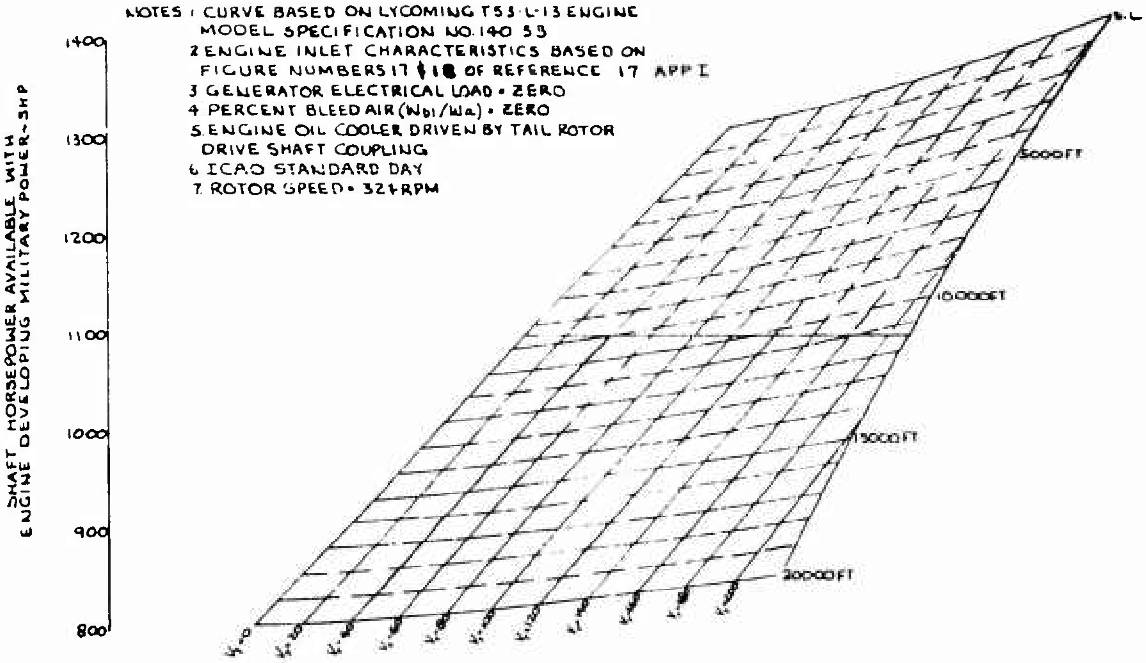


Figure No. 118

Specification Shaft Horsepower Available and Fuel Flow

AH 1G T53-L-13

NO ENGINE PARTICLE SEPARATOR INSTALLED



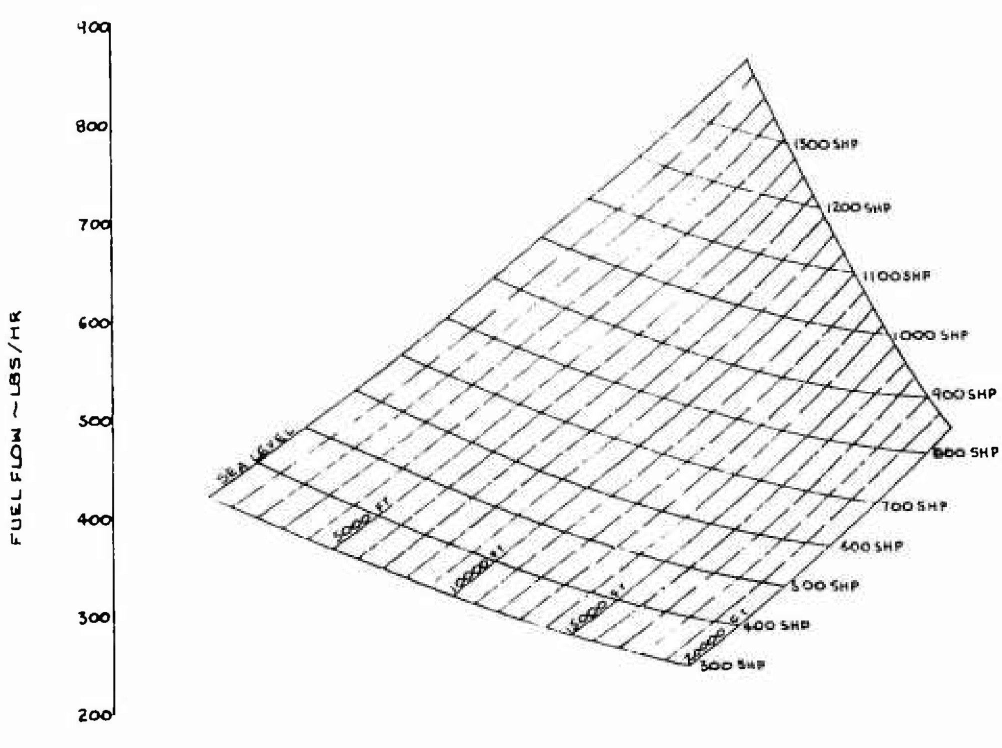


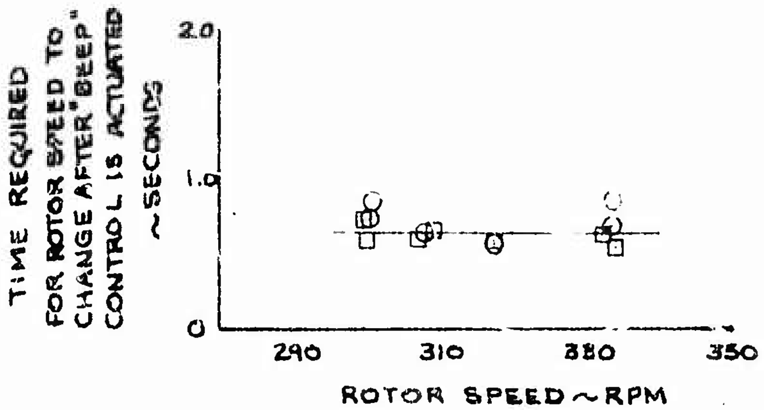
FIGURE NO. 119

ENGINE "BEEP" CONTROL CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

SYMBOL	AIRSPEED ~KCAS	DENSITY ALT. ~FT.	FLIGHT CONDITION
○	105	5000	LEVEL FLIGHT
□	ZERO	1500	GROUND RUN (COLLECTIVE FULL DOWN)



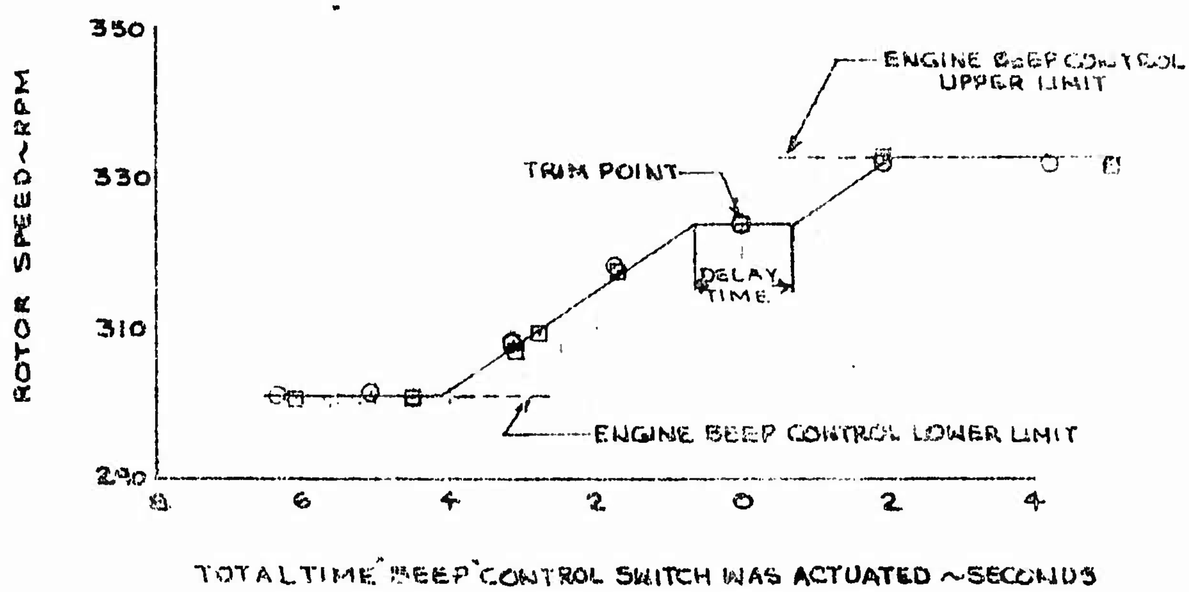


FIGURE NO 120

ENGINE RESPONSE TO A LEFT LATERAL CYCLIC CONTROL INPUT

AH-1G USA S/N 615247

T53-L-13 S/N LE 14001

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SCAS ON

GROSS WEIGHT = 8510 LB

CENTER OF GRAVITY = 190 INCHES (FWD)

COLLECTIVE CONTROL POSITION - 2.77 INCHES FROM FULL DOWN

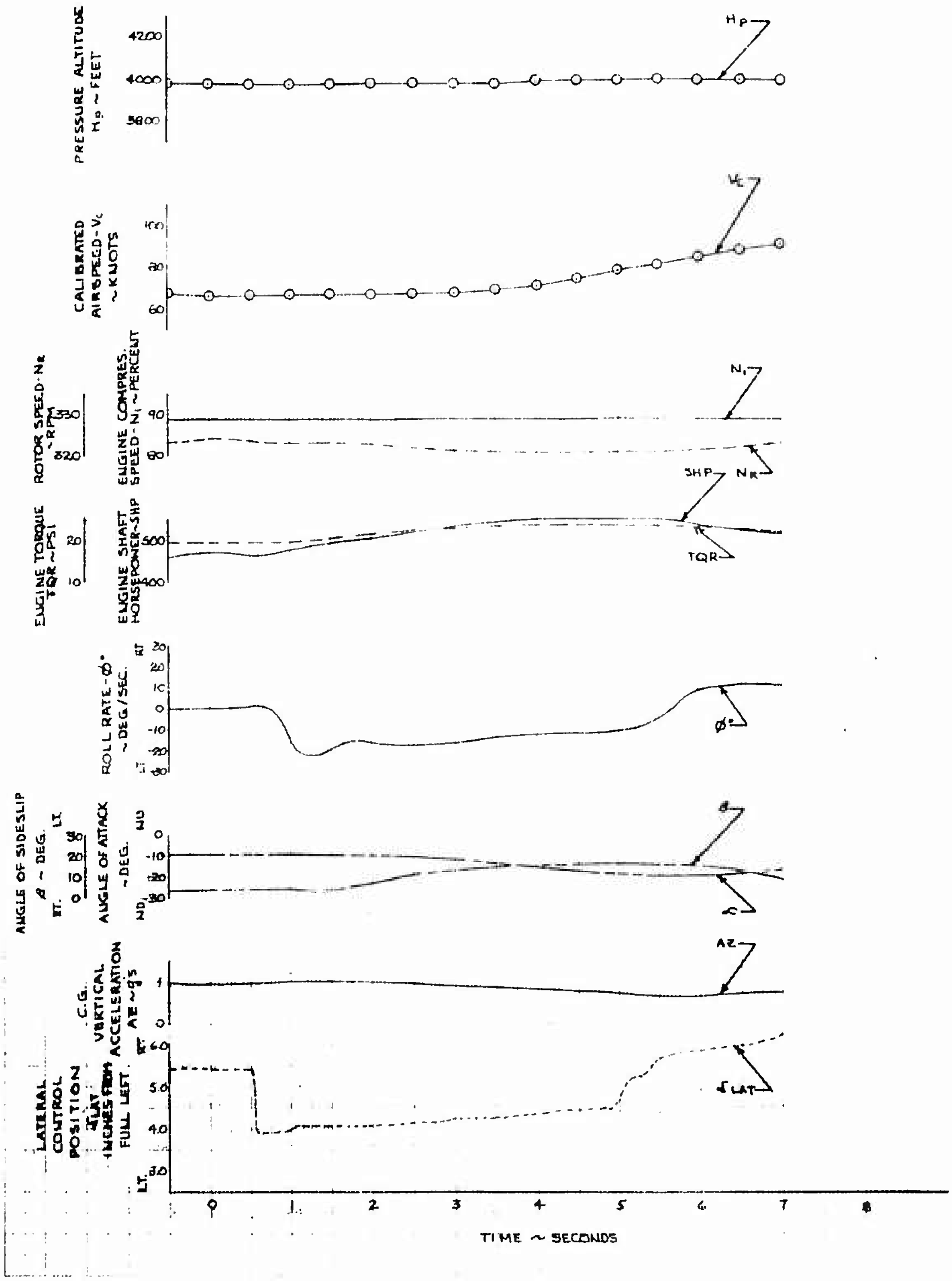


FIGURE NO 121

ENGINE RESPONSE TO A LEFT LATERAL CYLIC CONTROL INPUT

AH-1G USA 67-15247

T53-L-13/LE14001

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SCAS ON

GROSS WEIGHT = 8460 LB

CENTER OF GRAVITY = 190.0 IN (FWD)

COLLECTIVE CONTROL POSITION = 3.72 IN FROM FULL DOWN

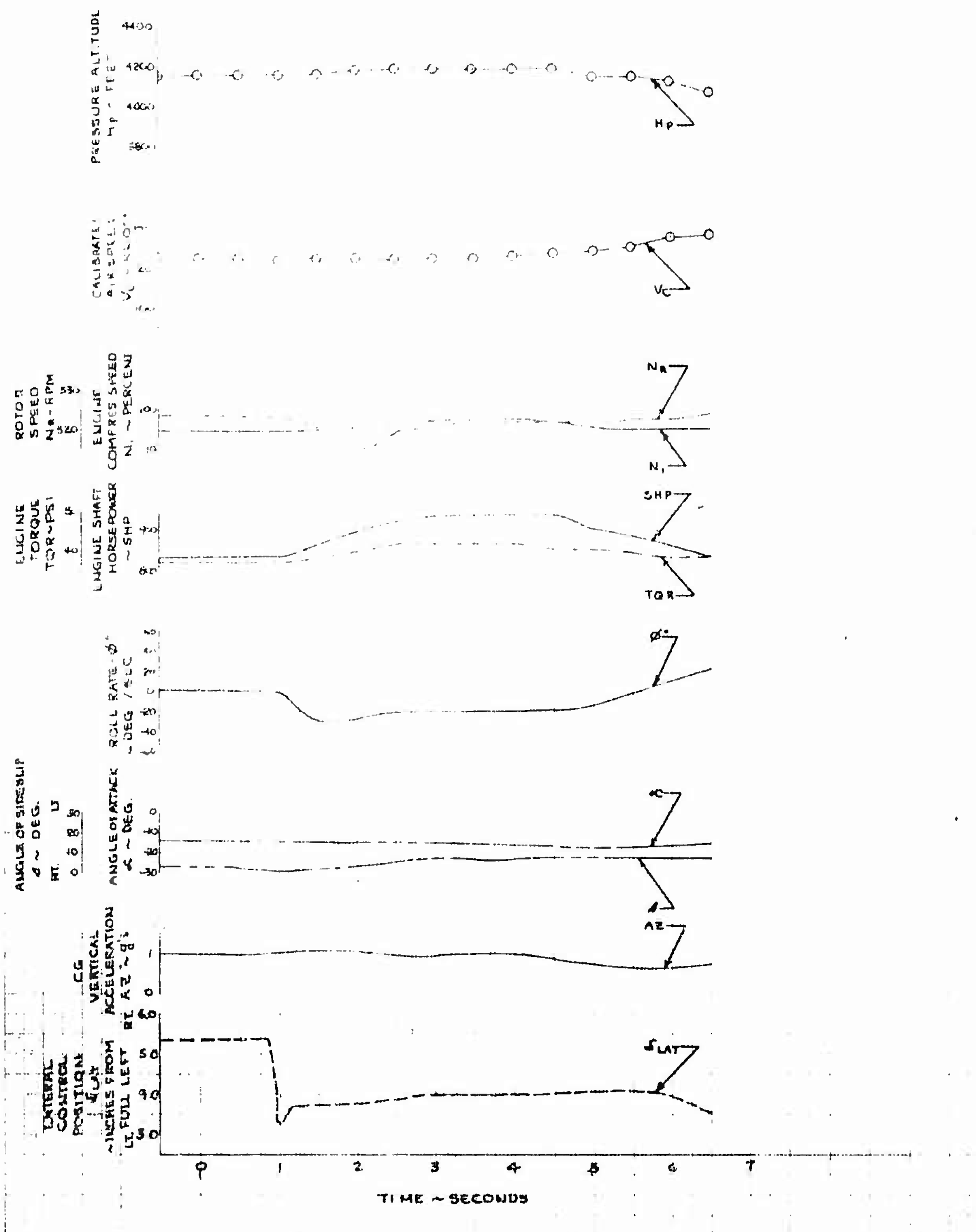


FIGURE NO 122

ENGINE RESPONSE TO A LEFT LATERAL CYCLIC CONTROL INPUT

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SCAS ON

GROSS WEIGHT - 8450 LB

CENTER OF GRAVITY - 190.0 IN (FWD)

COLLECTIVE CONTROL POSITION - 3.27 IN FROM FULL DOWN

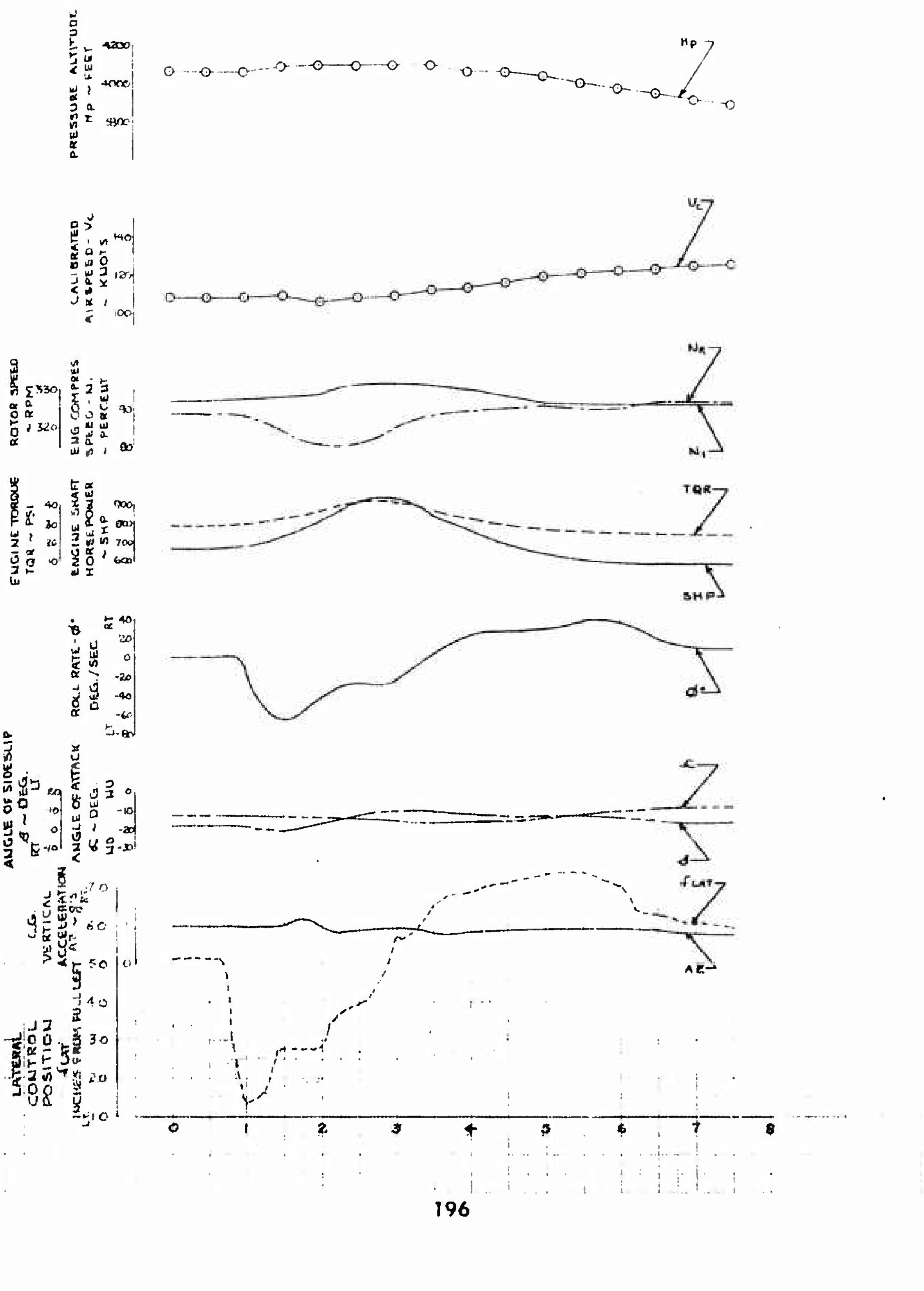


FIGURE No 123

ENGINE RESPONSE TO A RIGHT LATERAL CYCLIC CONTROL INPUT

AH-1D USA S/N 615247

T53-L-13 S/N LE14001

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SCAS ON

GROSS WEIGHT = 8520 LB

CENTER OF GRAVITY = 189.1 IN (FWD)

COLLECTIVE CONTROL POSITION = 2.77 IN FROM FULL DOWN

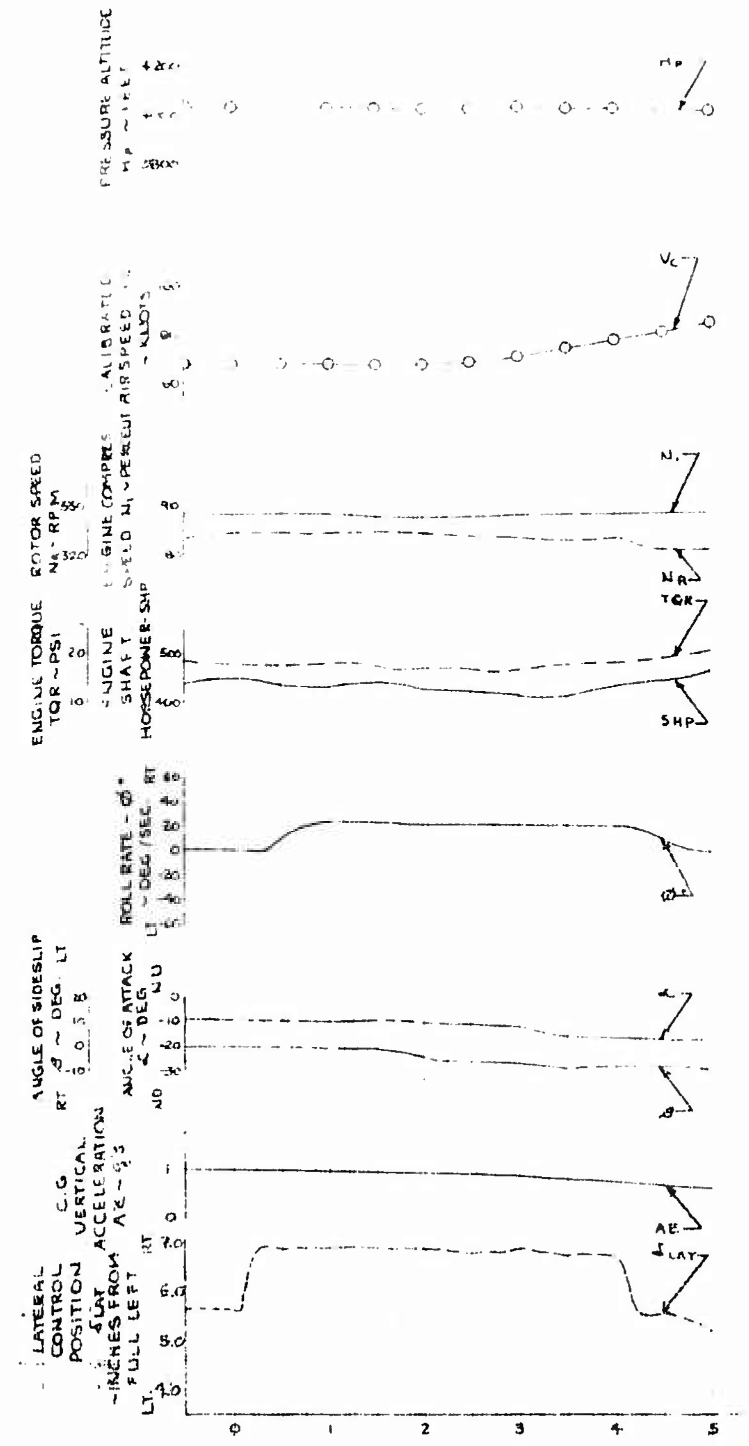


FIGURE NO 124

ENGINE RESPONSE TO A RIGHT LATERAL CYCLIC CONTROL INPUT

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HEAVY HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SCAS ON

GROSS WEIGHT - 8440 LB

CENTER OF GRAVITY - 190.0 IN (FWD)

COLLECTIVE CONTROL POSITION - 3.72 IN. FROM FULL DOWN

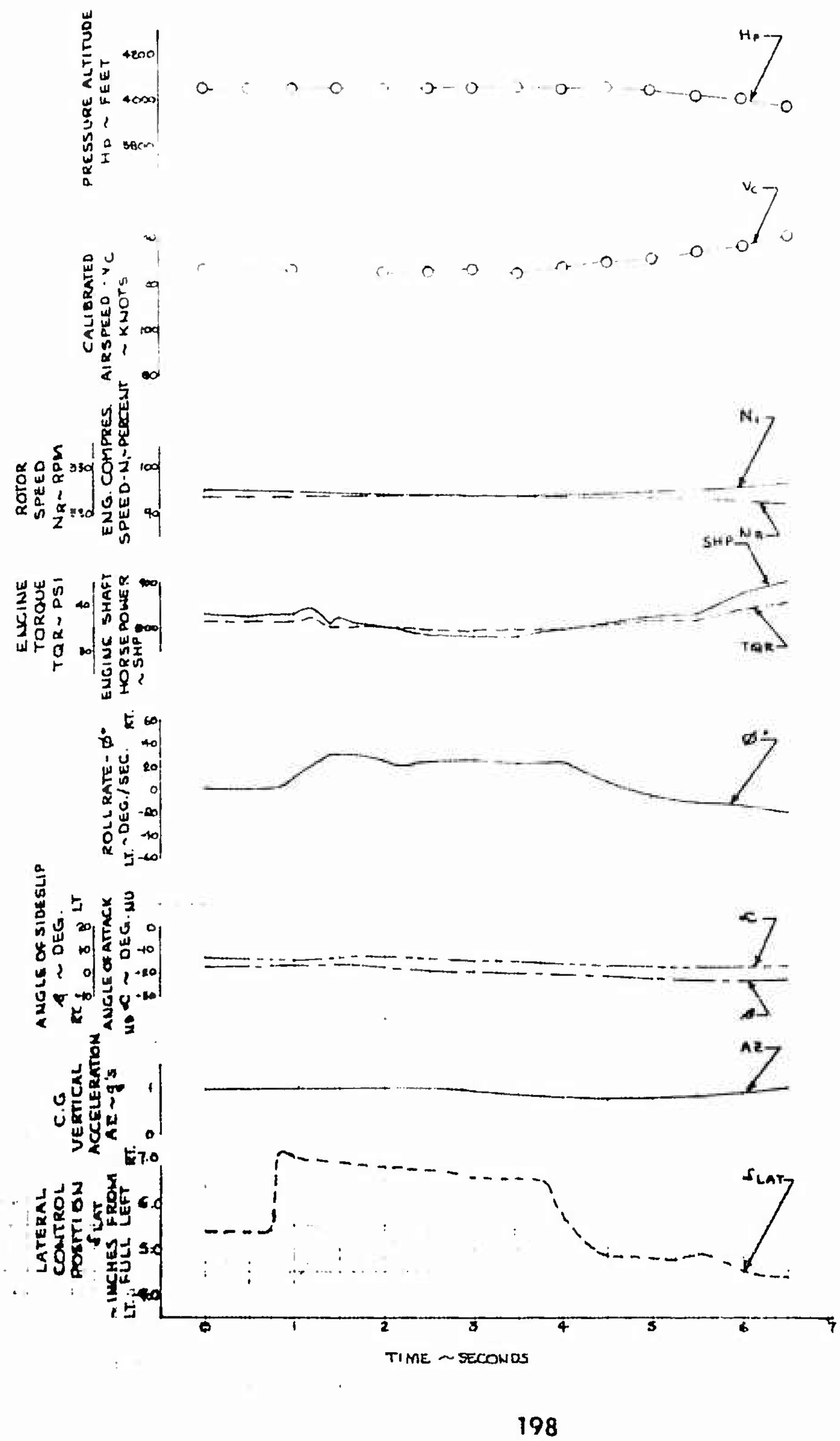


FIGURE NO. 125

ENGINE CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33

2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NO. 113 FOR ZERO AIRSPEED

3. GENERATOR ELECTRICAL LOAD = ZERO

4. PERCENT AIRBLEED (W_{bl}/W_a) = 0.067.

5. ROTOR SPEED = 324 RPM

6. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR

7. SOLID SYMBOLS DERIVED FROM CONTRACTOR'S CALIBRATED ENGINE DATA

SYM	DATE OF ENGINE CALIBRATION
■	22 AUGUST 1967
●	29 MARCH 1969

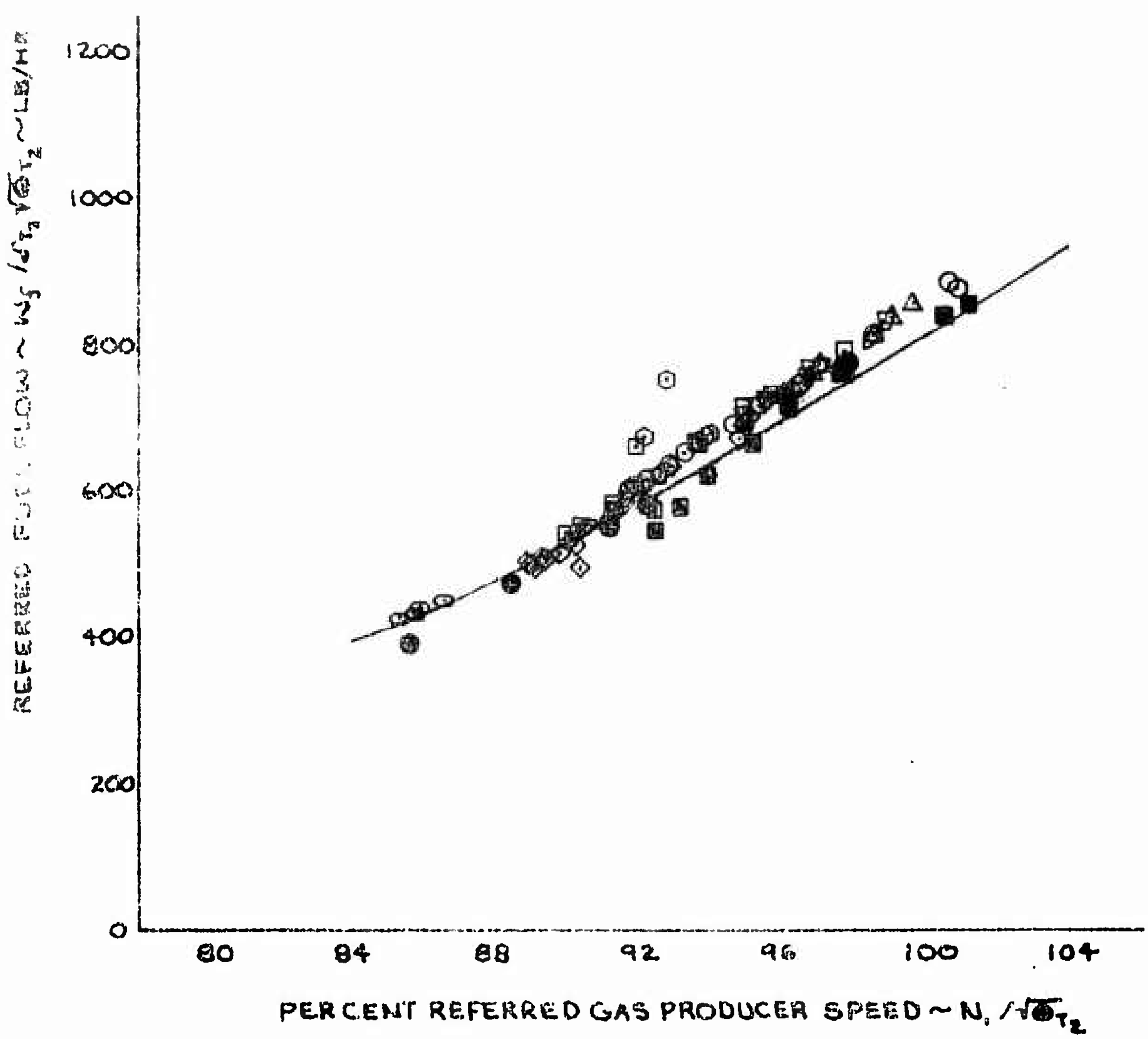


FIGURE NO 126

ENGINE CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES
1 CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 104 53
2 CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NO [illegible] FOR ZERO AIRSPEED
3 GENERATOR ELECTRICAL LOAD = ZERO
4 PERCENT AIRBLEED (W_{b1}/W_a) = 0 067
5 ROTOR SPEED = 324 RPM
6 ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR
7 SOLID SYMBOLS DERIVED FROM CONTRACTOR'S CALIBRATED ENGINE DATA

SYM	DATE OF ENGINE CALIBRATION
■	22 AUGUST 1967
●	29 MARCH 1969

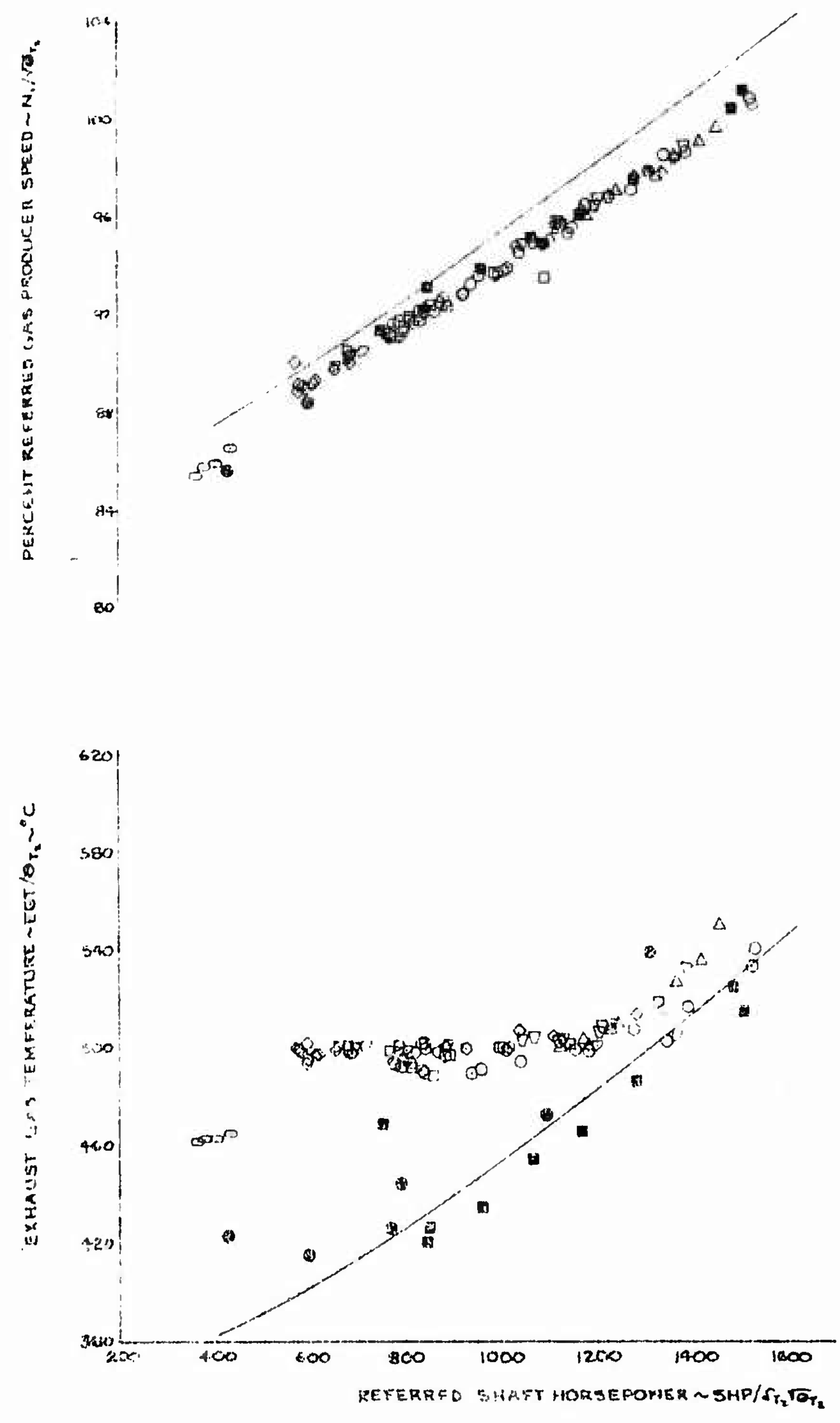


FIGURE NO. 21
ENGINE CHARACTERISTICS
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 104.33
2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NO 11 FOR ZERO AIRSPEED
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIRBLEED (W_{a1}/W_a) = 0.0%
5. ROTOR SPEED = 324 RPM
6. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR
7. SOLID SYMBOLS DERIVED FROM CONTRACTOR'S CALIBRATED ENGINE DATA

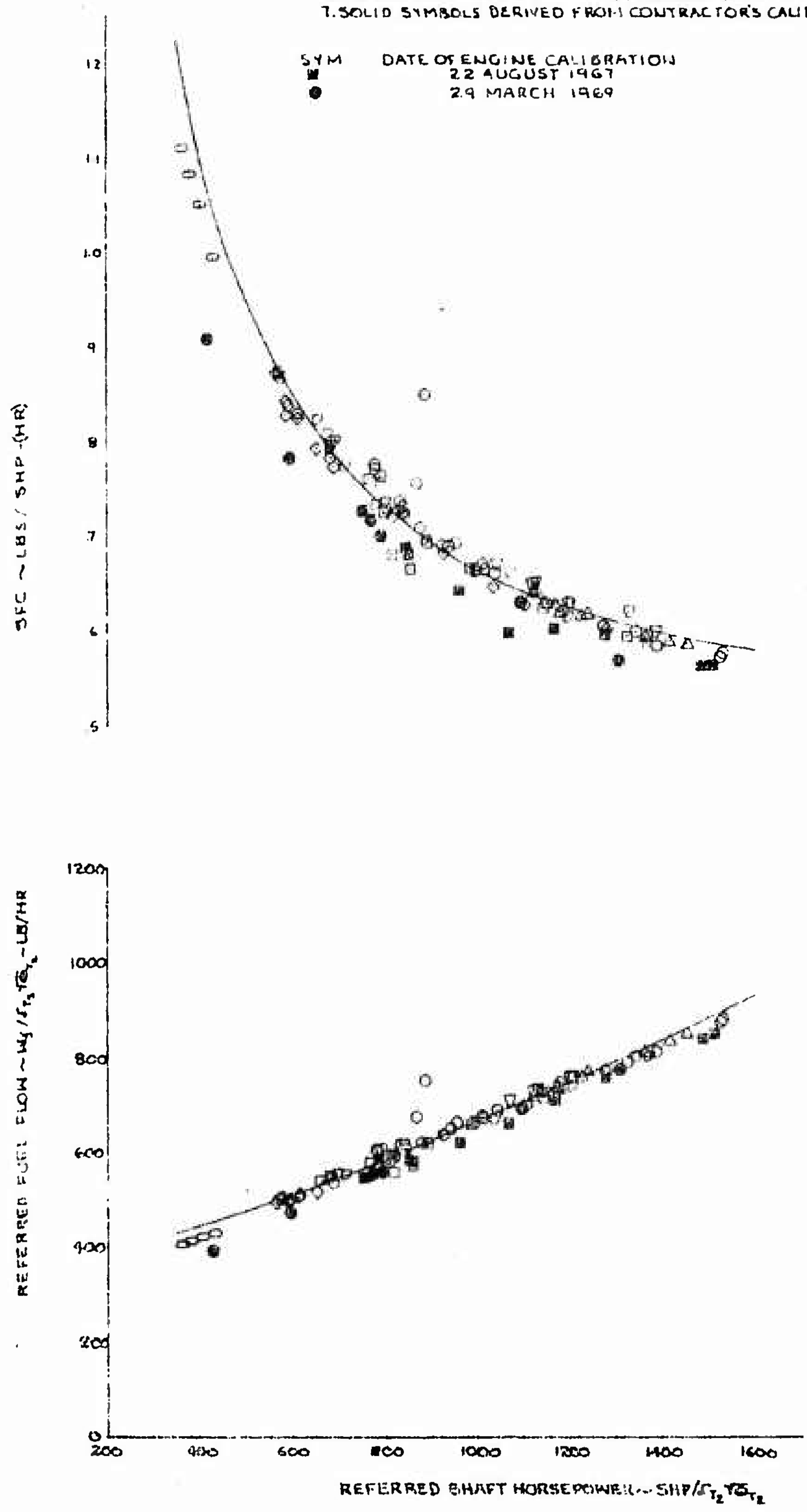


FIGURE NO. 128

ENGINE CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 S/N LE14008

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO.104.33

2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NO. 115 FOR ZERO AIRSPEED

3. GENERATOR ELECTRICAL LOAD = ZERO

4. PERCENT AIRBLEED (W_{bl}/W_a) = 0.06%

5. ROTOR SPEED = 324 RPM

6. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR

7. SOLID SYMBOLS DERIVED FROM CONTRACTOR'S CALIBRATED ENGINE DATA

SYM	DATE OF ENGINE CALIBRATION
●	5 JAN. 1968

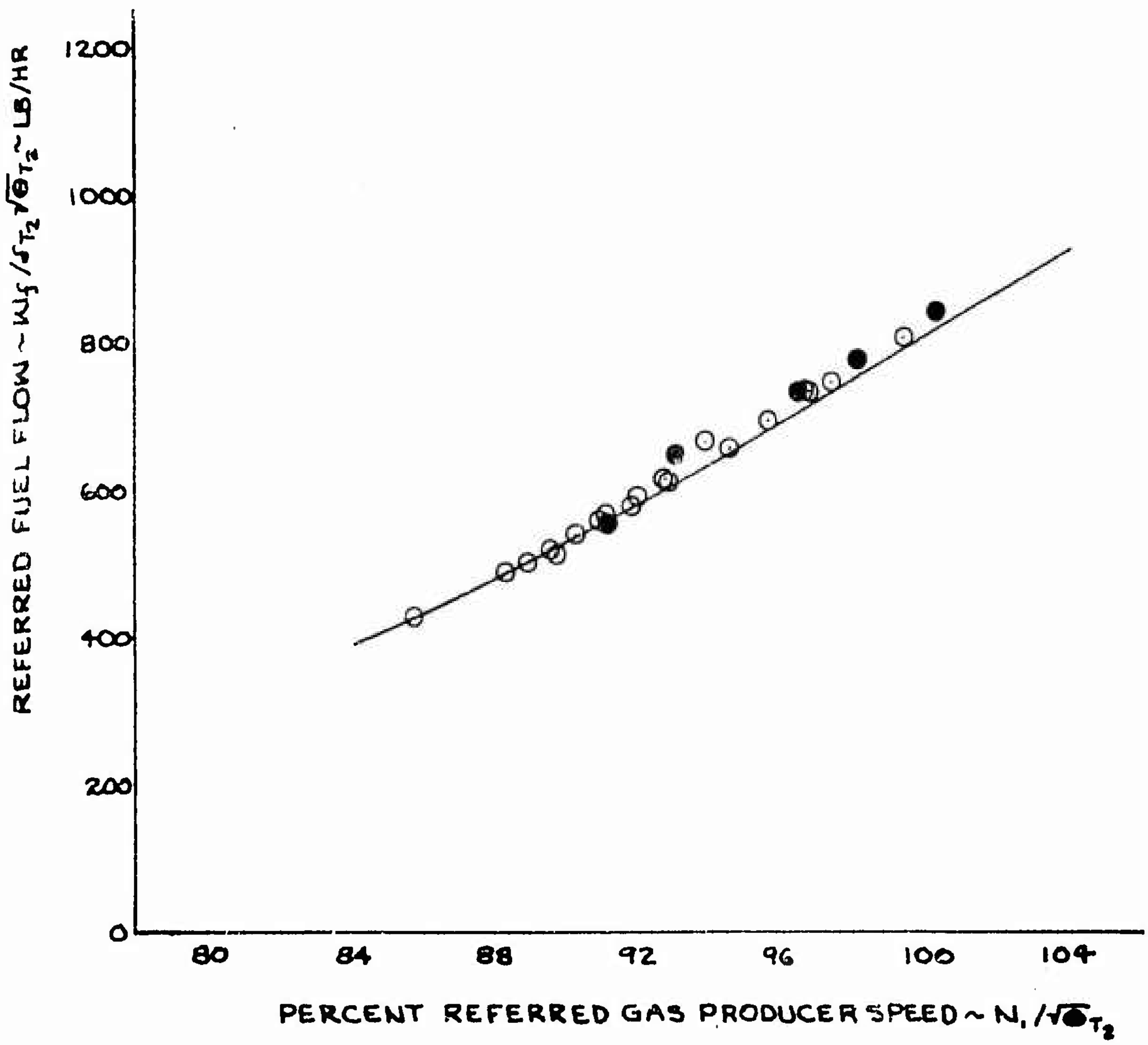


FIGURE No 129
ENGINE CHARACTERISTICS
AH-1G USA S/N 615247
T53-L-13 S/N LE14008
ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 104.33
2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NO. ... FOR ZERO AIRSPEED
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIRBLEED(W_{bl}/W_a) = 0.06%
5. ROTOR SPEED = 324 RPM
6. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR
7. SOLID SYMBOLS DERIVED FROM CONTRACTOR'S CALIBRATED ENGINE DATA

SYM	DATE OF ENGINE CALIBRATION
●	5 JAN 1968

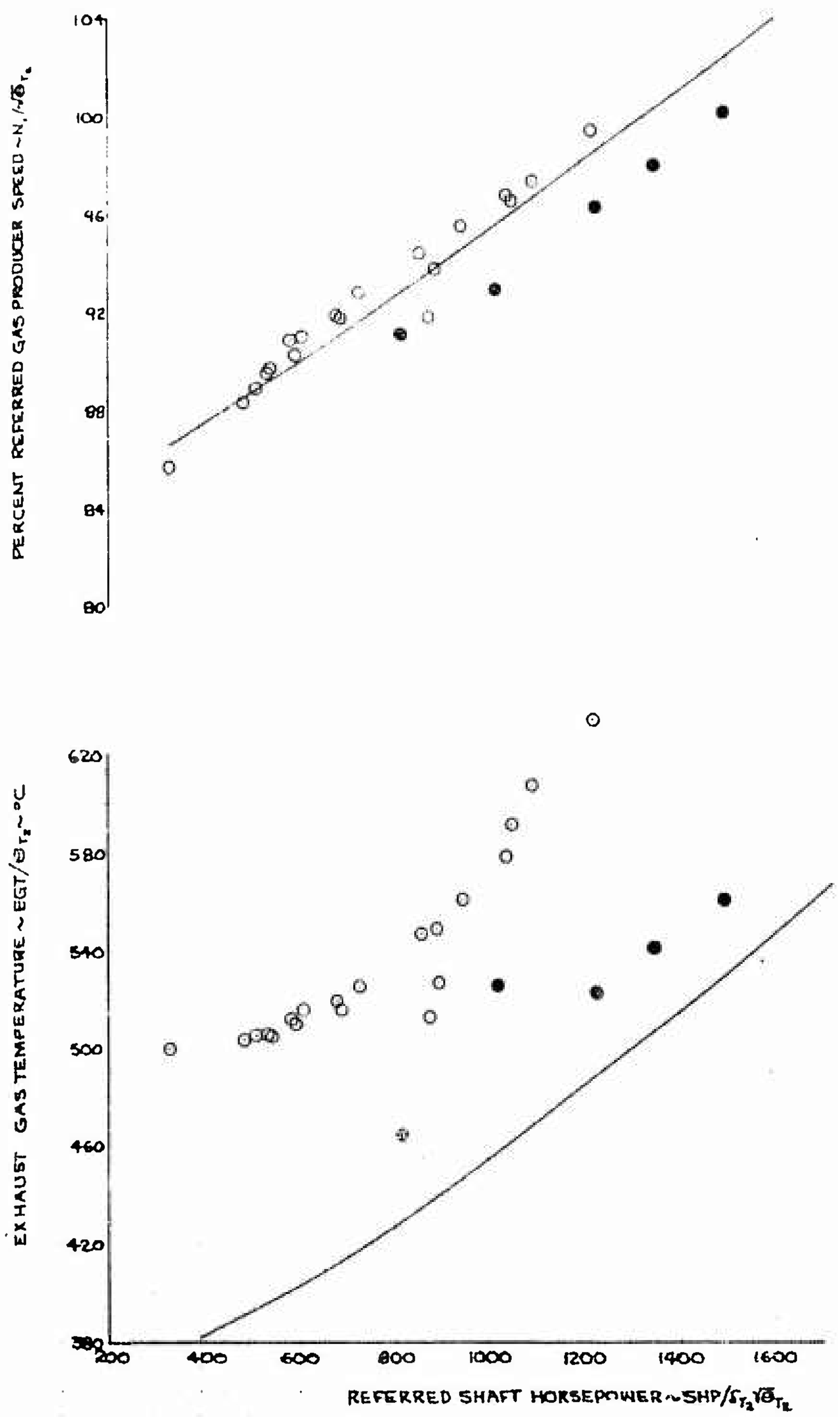


FIGURE No 130

ENGINE CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 LE14008

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES 1 CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO 104.33

2 CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NO FOR ZERO AIRSPEED

3 GENERATOR ELECTRICAL LOAD = ZERO

4 PERCENT AIRBLEED (W_{bl}/W_a) = 0.067

5 ROTOR SPEED = 324 RPM

6 ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR

7 SOLID SYMBOLS DERIVED FROM CONTRACTOR'S CALIBRATED ENGINE DATA

SYM DATE OF ENGINE CALIBRATION

● 5 JAN 1968

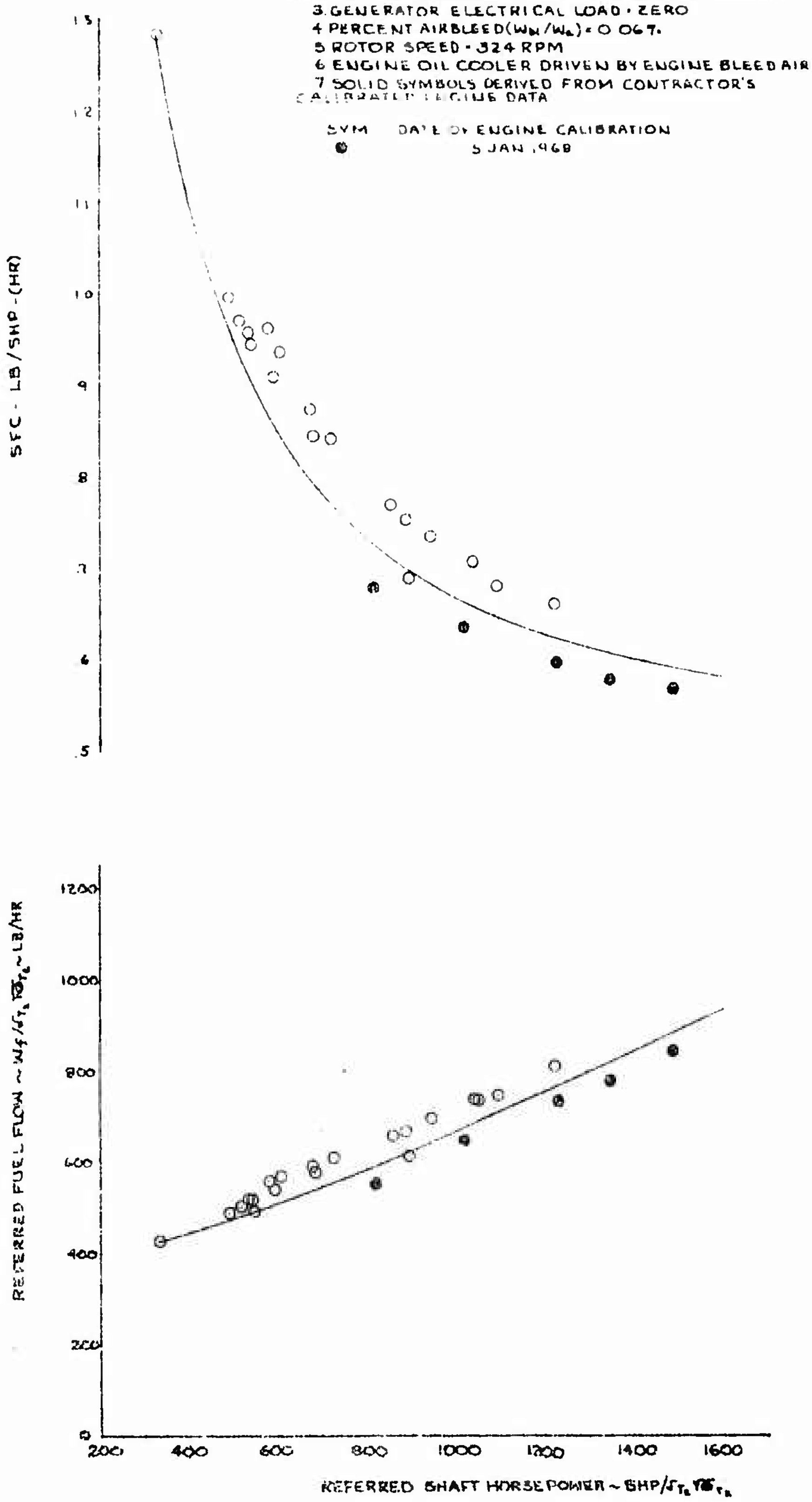


FIGURE NO. 131
AIRSPEED CALIBRATION
AH-1G T53-L-13
STANDARD AIRSPEED SYSTEM

SYM	AIRCRAFT S/N	CONFIGURATION	GR WT ~LB	DENSITY ALT. ~FT.	ROTOR SPEED ~RPM	LONG C.G. ~IN	SOURCE OF DATA
○	615247	CLEAN	7810	5500	324	191.7 (FWD)	PHASE D TEST PROG.
□	615246	BASIC	8280	5360	324	193.3 (FWD)	REF 2 APP. I
△	615248	BASIC	8170	4920	324	194.2 (MID)	REF 4 APP I
◇	615283	OUTB'D ALTERNATE	8290	3100	324	199.1 (AFT)	REF 5 APP I

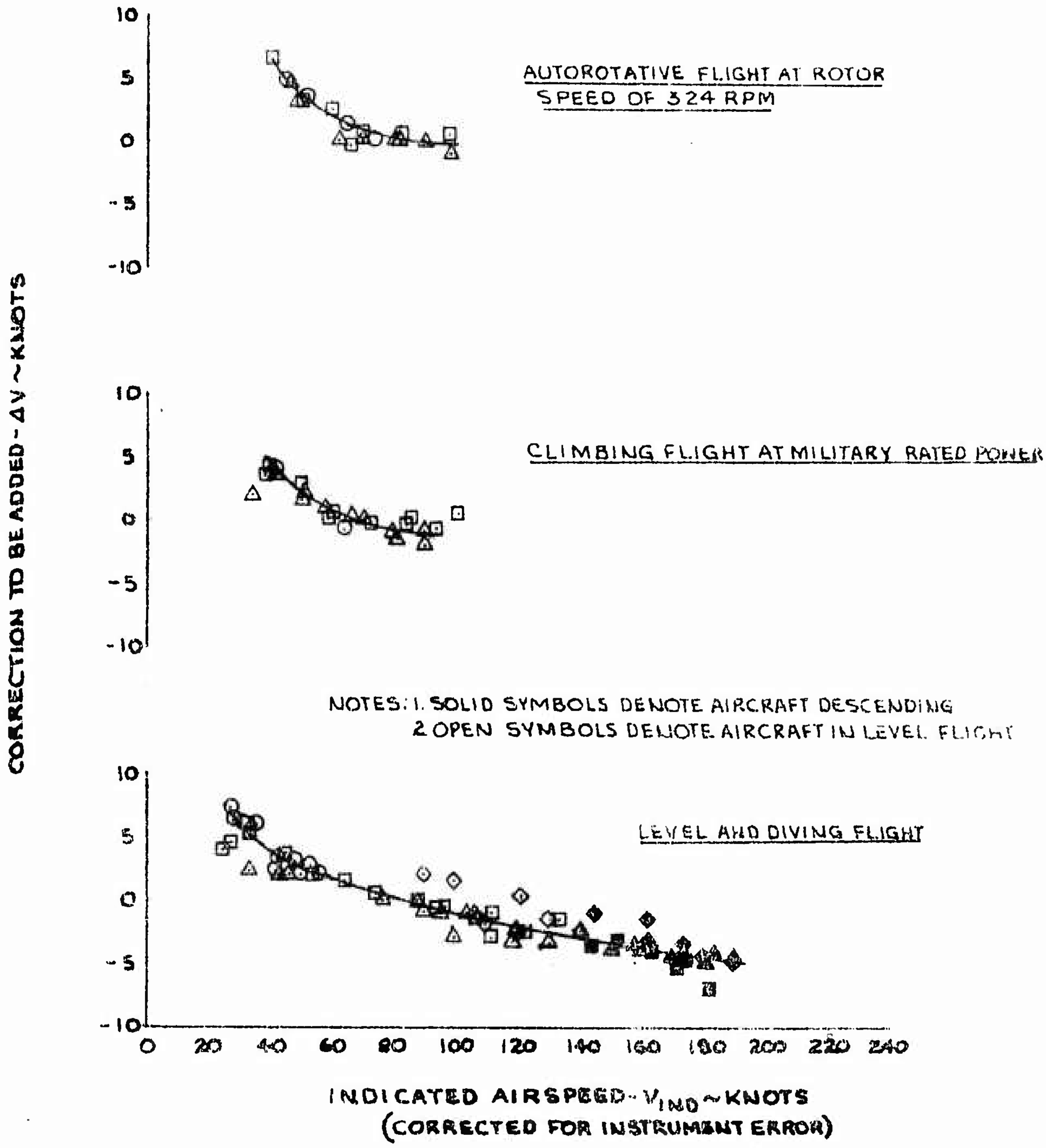
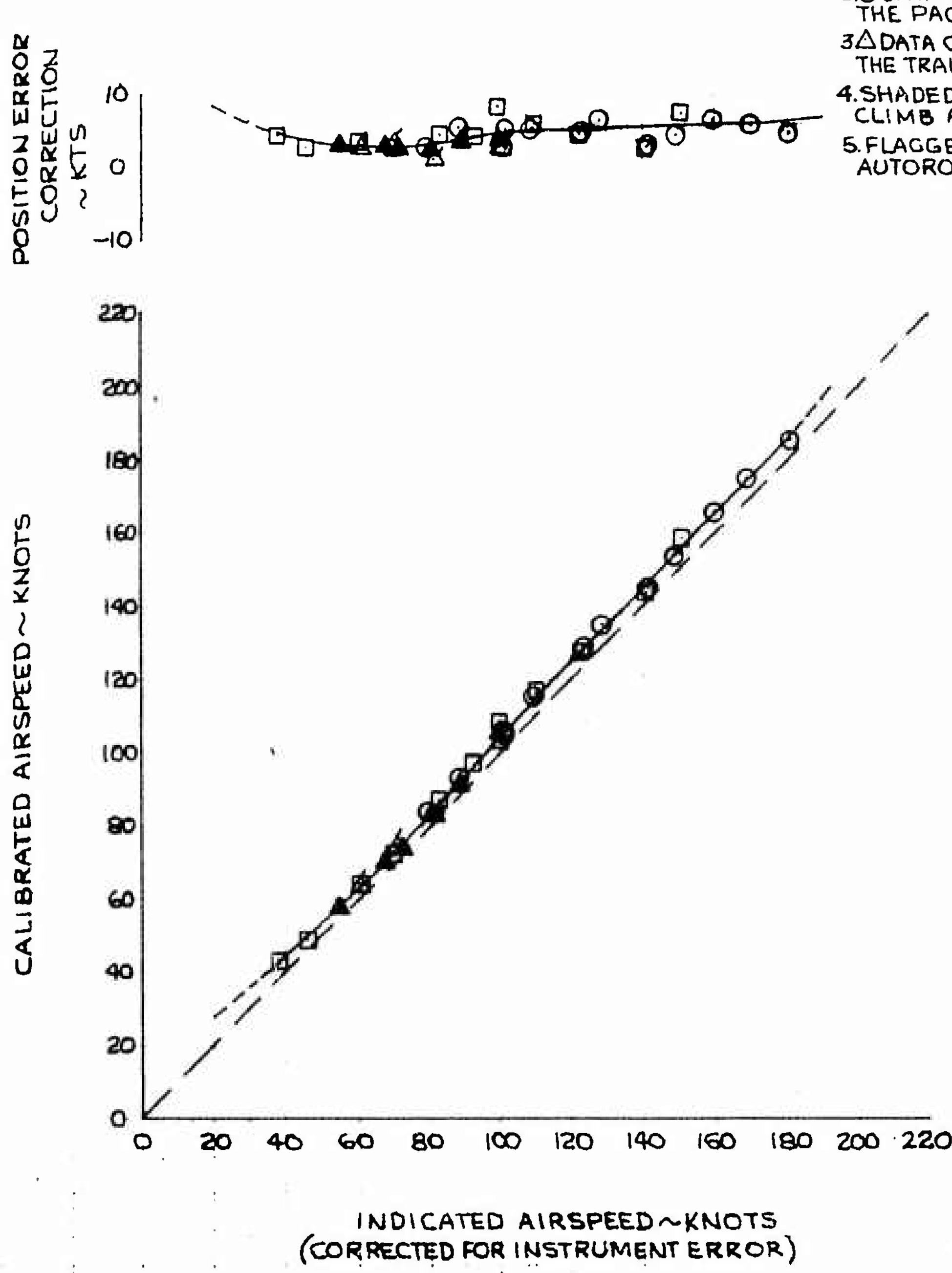


FIGURE NO 132
AIRSPEED CALIBRATION
AH-1G USA S/N 615247
BOOM SYSTEM

SYM.	GROSS WEIGHT ~LBS.	C.G STATION ~IN.	DENSITY ALTITUDE ~FT.	ROTOR SPEED ~RPM	CONFIGURATION
⊡	7265	193.5	1020 FT.	324	CLEAN
○	7176	193.3	5300 FT.	324	CLEAN
△	7200	193.3	5000 FT	324	CLEAN

NOTES: 1. ⊡ DATA COLLECTED USING THE GROUND SPEED METHOD.
2. ○ DATA COLLECTED USING THE PACER AIRCRAFT METHOD.
3. △ DATA COLLECTED USING THE TRAILING BOMB METHOD.
4. SHADED SYMBOLS DENOTE CLIMB AT LIMIT POWER
5. FLAGGED SYMBOLS DENOTE AUTOROTATION



APPENDIX VIII. SYMBOLS AND ABBREVIATIONS

Abbreviation	Definition	Unit
ALT	Altitude	foot
AVG	Average	--
CG, cg	Center of gravity	--
COND	Condition	--
CONF	Configuration	--
DEG, deg	Degrees	degree
DWN	Down	--
EGT	Engine exhaust gas temperature	°C
fig., figs.	Figure, figures	--
FLT	Flight	--
fpm	Feet per minute, foot per minute	ft/min
ft	Foot, feet	foot
FS	Fuselage station	inch
fwd	Forward	--
GRWT, grwt	Gross weight	pound
HQRS	Handling qualities rating scale	--
HR	Hour	hour
IFR	Instrument flight rules	--
IGE	In ground effect	--
in.	Inch, inches	inch
KCAS	Knots calibrated airspeed	knot

Abbreviation	Definition	Unit
KIAS	Knots indicated airspeed	knot
KTAS	Knots true airspeed	knot
LB, lb	Pound, pounds	pound
LT	Left	--
LONG.	Longitudinal	--
MAX, max	Maximum	--
MIN, min	Minimum	--
MRP	Military rated power	shp
NACA	National Advisory Committee for Aeronautics	--
NAMPP	Nautical air miles per pound of fuel	--
NAMT	Nautical air miles traveled	NM
ND	Nose down	--
NM	Nautical miles	--
NU	Nose up	--
NO., no.	Number	--
OGE	Out of ground effect	--
PSI, psi	Pound(s) per square inch	lb/in^2
ref	Reference, referred	--
RPM, rpm	Revolution(s) per minute	rpm
RT	Right	
SCAS	Stability and control augmentation system	--
SEC, sec	Second	--

Abbreviation	Definition	Unit
SFC	Specific fuel consumption	--
SHP, shp	Shaft horsepower	--
SL	Sea level	--
S/N	Serial number	--
STD, std	Standard	--
SYM	Symbol	--
TRQ	Engine output torque	in-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	--
dhp/dt	Rate of altitude change	ft/min
f	Equivalent flat plate area	ft^2
h	Skid height	foot
H_D	Density altitude	foot
H_P	Pressure altitude	foot
K_P	Engine power correction coefficient for climbing flight	--
K_W	Gross weight correction coefficient for climbing flight	--

Symbol	Definition	Unit
l_t	Distance from center line of main rotor shaft to center line of a 90-degree gear box output shaft	foot
M	Mach number	--
N_E	Engine speed	rpm
N_R	Main rotor speed	rpm
N_{TR}	Tail rotor speed	rpm
N_1	Engine compressor speed	percent
P	Engine output torque pressure	in. of Hg
R	Rotor radius	foot
R/C	Rate of climb	ft/min
R/D	Rate of descent	ft/min
s_g	Ground distance required to clear a 50-foot obstacle	foot
T	Temperature	°F, °C
V_{cal}	Calibrated airspeed	knot
V_{cruise}	Cruise airspeed	knot
V_H	Maximum airspeed for level flight	knot
V_L	Limit airspeed	knot
V_T	True airspeed	knot
W_a	Engine air flow	lb/hr
W_{bl}	Engine bleed air flow	lb/hr
W_f	Engine fuel flow	lb/hr

Symbol	Definition	Unit
°C	Degree(s) centigrade	degree
°F	Degree(s) Fahrenheit	degree
%	Percent	--
α	Angle of attack	degree
β	Angle of sideslip	degree
Δ	Difference	--
δ_{t_2}	Engine inlet pressure ratio	--
δ_{COLL}	Collective control position	inch
δ_{DIR}	Directional control position	inch
δ_{LAT}	Lateral cyclic control position	inch
δ_{LONG}	Longitudinal cyclic control position	inch
θ	Aircraft pitch attitude	degree
θ_{t_2}	Engine inlet temperature ratio	--
μ	Main rotor tip speed ratio	--
ρ	Air density	slugs/ft^3
σ	Density ratio	--
ϕ	Aircraft roll attitude	degree
$\phi°$	Aircraft roll rate	deg/sec
Ω	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

APPENDIX IX. DISTRIBUTION

Agency	Test Plans	Interim Reports	Final Reports
Commanding General US Army Aviation Systems Command			
ATTN: AMSAV-R-F	5	5	6
AMSAV-C-A	-	-	2
AMSAV-D-ZDOR	-	-	2
AMSAV-R-EH	1	1	2
AMSAV-R-R	-	-	1
PO Box 209 St. Louis, Missouri 63166			
Commanding General US Army Materiel Command			
ATTN: AMCPM-AAWS	5	1	5
PO Box 209 St. Louis, Missouri 63166			
Commanding General US Army Materiel Command			
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AMCMR	2	-	2
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Washington, D. C. 20315			
Commanding General US Army Combat Developments Command			
ATTN: USACDC LnO	11	11	11
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Fort Monroe, Virginia 23351			

Agency	Test Plans	Interim Reports	Final Reports
Commanding General US Army Test and Evaluation Command			
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USMC LnO	1	1	1
Aberdeen Proving Ground, Maryland 21005			
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SAVFE-AV	-	-	1
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Commanding General US Army Aviation Center Fort Rucker, Alabama 36362	1	1	1
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Commanding General US Army Electronics Command ATTN: AMSEL-VL-D Fort Monmouth, New Jersey 07703	-	-	1

Agency	Test Plans	Interim Reports	Final Reports
Commanding General US Army Weapons Command			
ATTN: AMSWE-RDT	-	-	2
AMSWE-REW	-	-	2
(Airborne Armament Flying) Rock Island Arsenal Rock Island, Illinois 61202			
Commandant US Marine Corps Washington, D. C. 20315	-	-	1
Director US Marine Corps Landing Force Development Center Quantico, Virginia 22133	1	1	2
US Air Force, Aeronautical Systems Division			
ATTN: ASNFD-10 Wright Patterson Air Force Base, Ohio 45433	-	-	1
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13. ABSTRACT

The Phase D, Part 2 airworthiness and qualification performance tests of the AH-1G helicopter were conducted in California at Edwards Air Force Base and auxiliary test sites during the period 13 June 1968 through 29 July 1969. Specific performance parameters were evaluated to determine model specification compliance and to obtain detailed performance and mission capability information for inclusion in technical manuals and other publications. The AH-1G exceeded all contractor performance guarantees. There were two deficiencies which affect the mission accomplishment of the helicopter: insufficient directional control which limits hovering, take-off and landing performance; and excessive tail rotor horsepower required for hovering flight. There were three shortcomings for which corrective action is desirable: the inability to achieve maximum tail rotor blade angle (19 degrees) when full left directional control is applied for all conditions with the present directional control/yaw SCAS geometry; excessive pilot effort required to maintain optimum climb and maximum endurance airspeeds; and the possibility of inadvertently exceeding the main transmission torque limit following a left-lateral control input when below the engine critical altitude.

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	ROLE	WT	ROLE	WT	ROLE	WT
AH-1G Helicopter						
Phase D, Part 2						
Performance						
Exceeded all contractor guarantees						
Two deficiencies						
Insufficient directional control						
Excessive tail rotor horsepower required						
Three shortcomings						
Tail rotor blade angle						
Excessive pilot effort required						
Main transmission torque limit						

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