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2 of 2

FLIGHT TEST PERFORMANCE PHASE - OH-6A HELICOPTER UNARMED (CLEAN) & ARMED WITH XM-7 OR XM-8 WEAPON SUBSYSTEM

ATA-TR-63-25-2

00188 ATA 63-25

U.S. ARMY

TEST & EVALUATION COMMAND

63-25 Part II

USATECOM PROJECT NO. 4-3-0250-51/52/53

PART TWO OF TWO PARTS

REPORT

OF THE ENGINEERING FLIGHT TEST

PERFORMANCE PHASE

OF THE

OH-6A HELICOPTER, UNARMED (CLEAN) and ARMED

WITH THE XM-7 or XM-8 WEAPON SUBSYSTEM

AUGUST 1964

Record Set

HEADQUARTERS

U.S. ARMY AVIATION TEST ACTIVITY

EDWARDS AFB, CALIFORNIA

Cancelled 17Feb71
Per AR 340.16 Par 3b
SAVTE-T

Cancelled (AR 345-15 Par 10)
US Army Aviation Systems Test Activity

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U. S. ARMY AVIATION TEST ACTIVITY

PART TWO OF TWO PARTS
REPORT
OF THE ENGINEERING FLIGHT TEST
--PERFORMANCE PHASE--
OH-6A HELICOPTER, UNARMED (CLEAN) AND ARMED
WITH THE XM-7 OR XM-8 WEAPON SUBSYSTEM

USATECOM PROJECT NO. 4-3-0250-51/52/53
ATA-TR-63-25-2
DA PROJECT NO. 1R141803D168

August 1964

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FOREWORD

Essential to an understanding of the results of aircraft testing is an understanding of the differences between engineering and service testing.

Engineering testing, using instrumented aircraft and calibrated instruments, can determine and record the exact performance, control response and limits, engine performance and power available, through accurate measurements and reduction of data to standard conditions. Thus, it is possible to determine when an aircraft is approaching or exceeding design limits or other specified criteria.

Service testing, using aircraft in standard configuration, results in a qualitative evaluation for user-type information. This information is based on a broad scope of pilot experience and technique provided by pilots ranging from those recently out of school to those with considerable field operational experience. The installed instruments and gauges are used to determine significant operating data. These instruments are not usually calibrated but represent typical instruments found in production helicopters. These instruments and gauges are verified for accuracy within acceptable tolerances but do not attain the precision provided by the calibrated equipment used for engineering testing.

The service test-pilot makes qualitative observations on only what he experiences during normal service flying. These observations are not correlated to such factors as the margin of control remaining or exact rates of control response. Exact measurements of such factors are necessarily the responsibility of the engineering test agency. Thus, service testing may show that the aircraft is suitable for performing a mission when, actually, flight has been performed close to, or within, control margins specified by military specifications. What may appear to be discrepancies between service and engineering test reports is actually the difference between qualitative and quantitative reporting.

The Light Observation Helicopter evaluation is the first combined aircraft engineering and service test program that has resulted in coordination of reports and comparison of reports prior to procurement decision. Caution must be exercised, therefore, to preclude taking an item out of context in any one report to establish a particular position. Seeming inconsistencies can be reconciled only by examination of all reports with due regard to the specific conditions under which the test was accomplished.

ABSTRACT

Performance tests were conducted on the OH-6A to determine the compliance with the Light Observation Helicopter (LOH) Military Characteristics and to check the contractual guarantees outlined in the OH-6A Model Specification.

The U. S. Army Aviation Test Activity (USAATA), Edwards Air Force Base, California, was designated Executive Test Agency for the Confirmatory Engineering Tests in the LOH Program and is responsible for test execution and test reporting of its assigned phase.

Engineering flight tests were conducted by the U. S. Army Aviation Test Activity at Edwards Air Force Base, California, and at auxiliary test sites at Bakersfield and Bishop, California. A total productive test time of 61:30 hours were flown during the period of 8 April to 1 July 1964.

All Model Specification performance guarantees and requirements stated in the LOH Military Characteristics were evaluated using power available and fuel flow characteristics as defined in Engine Model Specification No. 580-A.

The OH-6A met the contractual performance guarantee for maximum airspeed.

The OH-6A did not meet the contractual performance guarantees for endurance or for 35 degrees Centigrade (C) out-of-ground effect (OGE) hovering ceiling.

The OH-6A met the Military Characteristics requirements for 3 hours endurance with a 400 pound payload, cruise speed and endurance at 85 percent cruise power.

The OH-6A did not meet the Military Characteristics requirements for OGE hot day hover performance at 6000 feet or OGE hover performance at overload gross weight, sea level, standard day.

The OH-6A performance was summarized using power available as defined by Engine Model Specification No. 580-E.

The takeoff performance of the OH-6A was satisfactory under all gross weight and ambient atmospheric conditions that allowed a minimum in-ground effect (IGE) hover of 2 feet. Under the ambient temperature and altitude conditions tested, the OH-6A did not possess an adequate hovering capability at overload gross weight to permit a takeoff evaluation. Climb and autorotation performance was considered satisfactory for a helicopter of this power and weight class.

This program was essentially an airframe evaluation, but a section has been incorporated into Appendix II in order to describe pertinent characteristics of the T63-A-5 engine.

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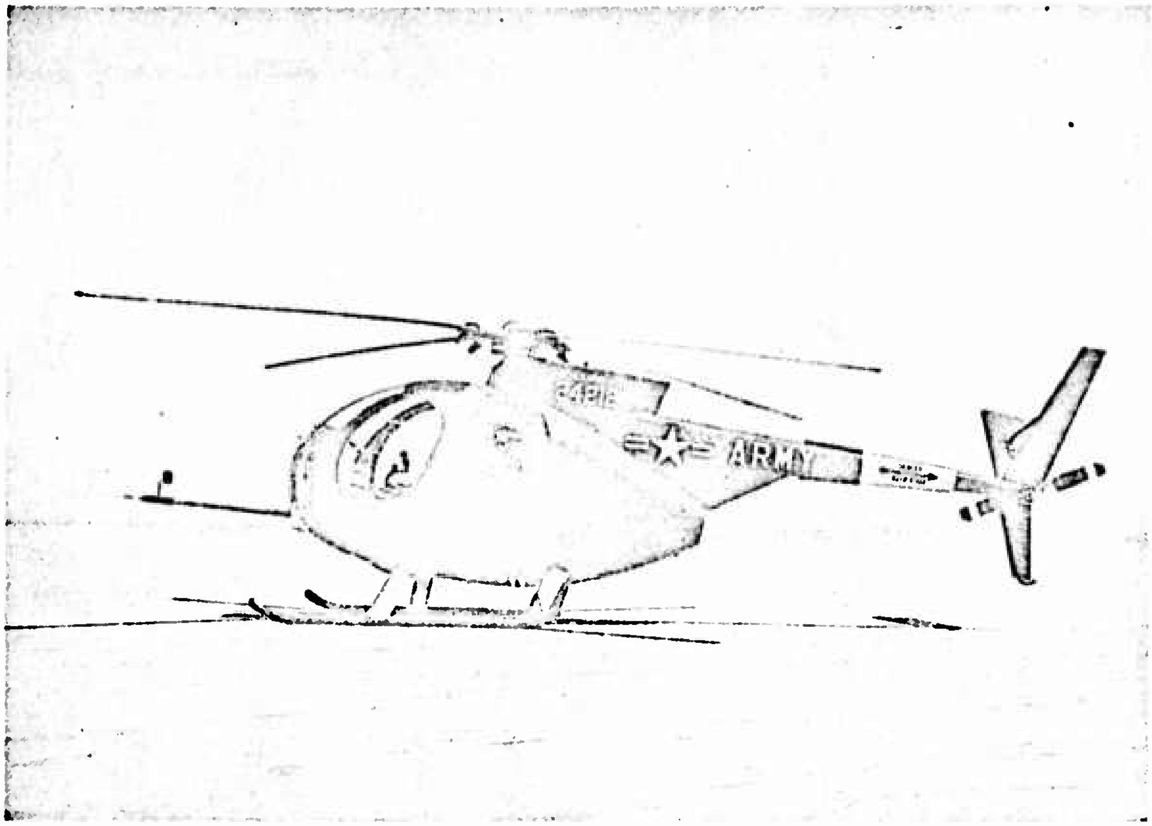


PHOTO 1 - OH-6A

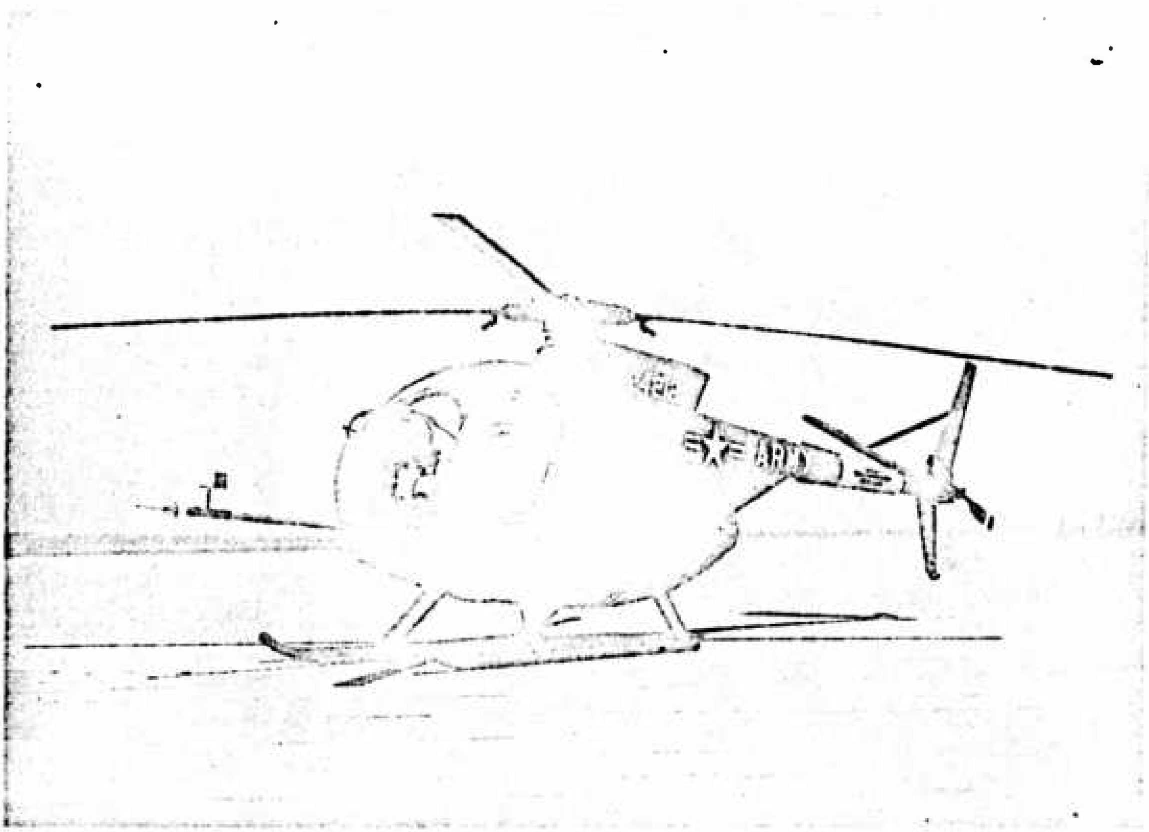


PHOTO 2 - OH-6A

PHOTO 3 – XM-7 ARMAMENT KIT

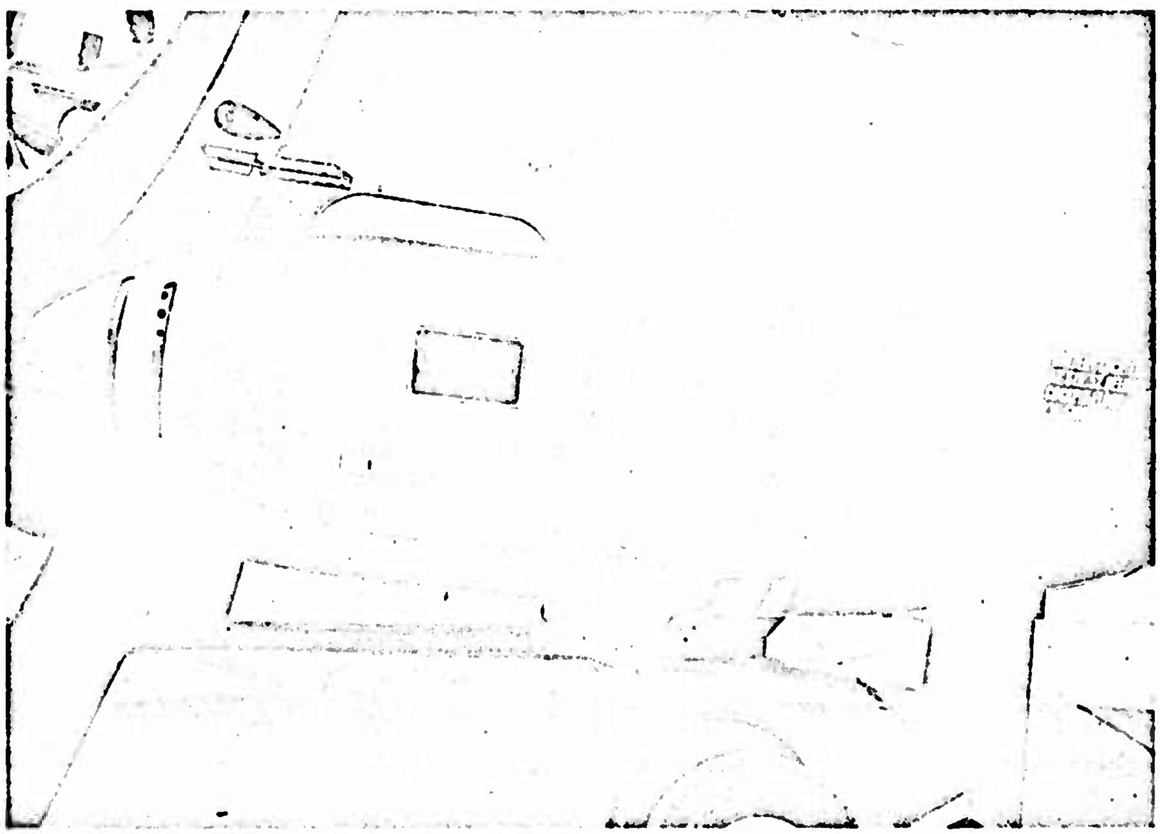


PHOTO 4 – XM-8 ARMAMENT KIT

SECTION 1 - GENERAL

1.1 REFERENCES

See Part I, Section 1, References a. to i., inclusive, and the following additional references:

j. Model Specification, Light Observation Aircraft, OH-6A, 26 July 1961.

k. Model Specification No. 580-A, Model T63-A Engine, 12 December 1960.

l. Model Specification No. 580-E, Model T63-A-5 Engine, 24 June 1963.

m. Letter, SMOSM-PAIA-2, Headquarters, U. S. Army Aviation Materiel Command, 4 April 1964, subject: "Compliance Check of Manufacturer's Guaranteed Performance and Competitive Performance Evaluation."

1.2 AUTHORITY

See Part I, Section 1.

1.3 OBJECTIVES

The objective of this program was to conduct engineering performance flight tests of the Light Observation Helicopter (LOH) Prototype OH-6A to; (a) confirm contractor compliance with the approved Army Military Characteristics for an unarmed (clean) and armed OH-6A helicopter; (b) provide data to assist in selecting an LOH design for possible future production; and (c) determine if the contractor performance guarantees were met.

1.4 RESPONSIBILITIES

See Part I, Section 1.

1.5 DESCRIPTION OF MATERIEL

See Part I, Section 1.

1.6 BACKGROUND

See Part I, Section 1.

1.7 FINDINGS

The takeoff performance was satisfactory. For the conditions tested, the minimum distance required to clear a 50-foot obstacle occurred at 20 knots true airspeed (KTAS). The optimum airspeed would, however, be less than 20 KTAS under some conditions.

The hovering performance of the OH-6A was satisfactory at sea level standard day conditions but deteriorated rapidly with altitude and increasing ambient temperature. The OH-6A did not meet the guaranteed out-of-ground effect (OGE) hovering performance, i.e, 6000 feet on a 95 degree Fahrenheit (F) day. Based on the engine power available presented in the T63-A engine Model Specification No. 580-A and the power required to hover OGE determined during these tests, the OH-6A could hover only at 3800 feet at 95 degrees F at design gross weight (2085 pounds). Using the same basis for power available, the maximum gross weight at which the helicopter could hover OGE on a sea level standard day was 2485 pounds. Using power available based on the T63-A-5 engine Model Specification 580-E, the OGE hovering ceiling at 95 degrees F was 3400 feet at design gross weight.

The climb performance of the OH-6A was satisfactory. The service ceilings (altitude at which the rate of climb is 100 feet/minute) were found to be 19,000 feet at 2085 pounds, and 11,500 feet at 2700 pounds using maximum continuous power. The aircraft exhibited undesirable pitch and yaw instabilities at the optimum climb airspeeds which made it difficult to obtain optimum climb performance.

In level flight, the OH-6A met the Vmax guarantee of 123 knots ± 10 percent (135.5 to 110.5 knots) at 2085 pounds sea level standard day at maximum continuous power. The maximum speed achieved at the above conditions was 120 knots.

The helicopter did not meet the endurance guarantee of 3.4 hours at a cruise speed of 100 knots, design gross weight, and sea level standard day. The maximum endurance under the above conditions was 3.02 hours. Endurance, payload, and cruise speed performance complied with the Military Characteristics performance requirements.

Even though the endurance guarantee was not met, the level flight performance of the OH-6A was considered satisfactory at gross weights of 2100 pounds or less and altitudes less than 5000 feet. The maximum airspeed was limited by power available at gross weights of 2100 pounds or less. At overload gross weights and altitudes in excess of 5000 feet, the maximum airspeed was limited by never-exceed airspeed (Vne).

The addition of either the XM-7 or the XM-8 armament kits reduced the airspeed at maximum continuous power by less than 5 knots true airspeed (KTAS).

During autorotation, the airspeed for minimum rate of descent was determined to be 53 knots calibrated airspeed (KCAS) at 469 rpm. The rotor rpm for minimum rate of descent could not be determined because of the high power turbine speed when the gas producer was at idle. Test results indicate that the minimum rate of descent would occur at a rotor speed below 440 rpm.

The test item satisfied all the performance requirements of the Military Characteristics except the 6000 feet, 95 degree F OGE hovering ceiling at design gross weight. On a sea level standard day the helicopter will not hover OGE at the OH-6A Model Specification overload gross weight (2700 pounds).

1.8 CONCLUSIONS

None

1.9 RECOMMENDATIONS

None

SECTION 2 - DETAILS AND RESULTS OF SUB-TESTS

2.0 INTRODUCTION

Performance flight tests on the OH-6A Light Observation Helicopter were conducted by the U. S. Army Aviation Test Activity at Edwards Air Force Base, California. Sea level and high altitude testing was accomplished at Bakersfield and Bishop, California, where altitudes from sea level to 9500 feet and a wide range of ambient temperatures were available. A total of 95 flights were conducted for 61.5 productive flight hours. The tests were accomplished during the period of 8 April to 1 July 1964.

Level flight performance data were obtained at Edwards Air Force Base and a sea level site at Bakersfield, California. Climb performance data were obtained entirely at Bakersfield, where climbs could be initiated near sea level. Autorotational descent performance data were collected during descents after completing other tests. Takeoff tests were accomplished in the Bishop, California area where pressure altitudes of 4100 feet (Bishop Airport) and 9500 feet (Coyote Flats) were available. Hovering performance data were collected at all of the test sites.

All tests were conducted in stabilized non-turbulent air so that accurate performance data could be obtained. The test data were recorded by hand from sensitive instruments or recorded automatically, using a photo panel.

Power available and fuel flow as specified in the T63-A engine Model Specification No. 580-A were used to check the contractual guarantees and the desired performance as defined in Military Characteristics, Light Observation Aircraft (See Part I, Section 1.1.a).

Summary performance was obtained using power available and fuel flow information based on the T63-A-5 engine Model Specification No. 580-E referred characteristics as defined on a sea level standard day. Engine performance at the 6000 foot altitude, 95 degree Fahrenheit (F) ambient temperature and all other altitude temperature combinations were obtained from curves of standard power deterioration with increasing altitude and temperature. These curves were derived using standard engineering methods and were verified with test stand engine calibration data on five different T63-A-5 engines. An engine that will produce the minimum 580-E specification sea level standard day power, will not produce the 206 Shaft Horsepower (SHP) at 6000 feet, 95 degrees F as shown in Model Specification No. 580-E. Such an engine will, in fact, produce only 199 SHP under these altitude, ambient temperature conditions (See Section 3, Appendix II, "General Aircraft Information," Model Specification 580-E Inaccuracies).

2.1 TAKEOFFS

2.1.1 OBJECTIVE

Takeoff tests were conducted to determine the performance of the OH-6A helicopter under conditions in which a vertical takeoff could not be made to clear a 50-foot obstacle. Under these conditions a short acceleration close to the ground will enable the aircraft to operate out of relatively short fields.

2.1.2 METHOD

Takeoff tests were conducted to obtain curves of climb-out airspeed versus distance required to clear 50 feet. Each curve was obtained by conducting a series of takeoffs using various climb-out airspeeds. During each series, ballast was added or removed as necessary so as to maintain the desired excess power available conditions as fuel was consumed and ambient temperature varied.

These tests were conducted at density altitudes of approximately 4200 feet and 10,000 feet. Gross weight was varied from 1790 to 2050 pounds at a mid C.G. location (Station 100). All takeoff tests were conducted with a main rotor speed of 469 rpm (100 percent power turbine speed (N_2)) using takeoff power. Takeoff performance was evaluated using only one takeoff technique since time did not permit use of a second technique. A ground operated Fairchild Flight Analyzer was used to produce a photographic record of time, horizontal distance, and vertical distance for each takeoff.

The climb-out airspeed range used for each series of takeoffs varied from the minimum achievable to maximum practical airspeed (approximately 60 knots indicated airspeed (KIAS)). All takeoff tests were performed in winds of 2 knots or less.

2.1.3 RESULTS

Test results are presented graphically in Figures No. 4 through 8 and are summarized in Figures No. 1 ahd 3, Section 3, Appendix I.

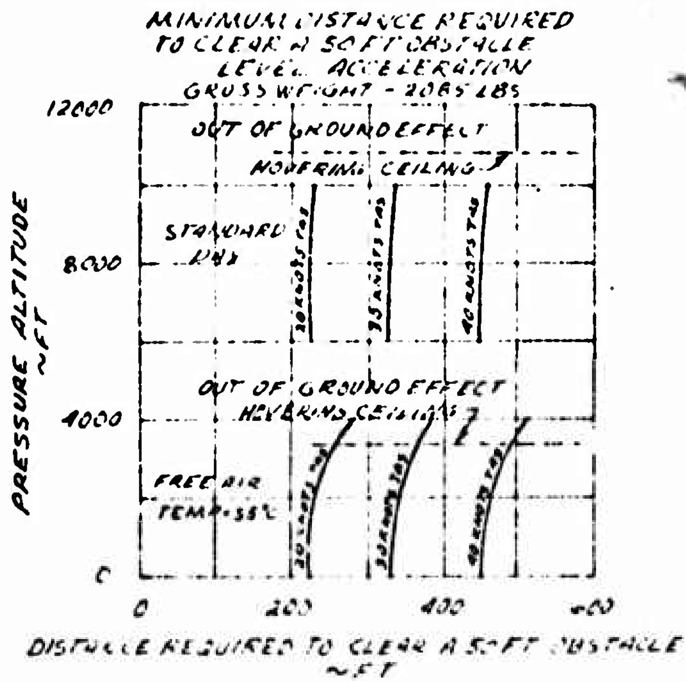
2.1.4 ANALYSIS

a. Comparative Techniques

The summary takeoff performance curves presented in Figure A on page 6 show the variations of takeoff distance with pressure altitude and climb-out airspeed. These curves are presented for climb-out true airspeeds of 20, 30 and 40 knots under both standard day and hot day (35 degrees Centigrade (C)) conditions at a gross weight of 2085 pounds. These curves were derived from Figures No. 4 and 16, Section 3, Appendix I. Takeoff distances obtained at conditions where

OGE hover was possible, show little variation in distance with altitude. When OGE hover was possible, the helicopter could be climbed vertically hence minimum takeoff distance was essentially zero. In Figure A, below, the distances required for takeoff when the OGE hover capability was present are essentially the distance required to accelerate to the particular climb speed.

FIGURE A



It would appear from the above curves that 20 knots true airspeed (KTAS) was the optimum climb-out airspeed in all cases. Actually, the optimum climb airspeed varies depending upon power available and under some conditions will be less than 20 KTAS. The limited calendar time available for the conduct of these tests prevented the determination of optimum climb-out airspeeds for all conditions.

Summary takeoff performance was not presented for hot day (35 degrees Centigrade (C)) overload gross weight (2700 pounds) conditions. Under these conditions, the helicopter did not possess an adequate hovering capability to enable a presentation to be made. It is doubtful if successful takeoffs could be made consistently at overload gross weight at no wind, sea level, hot day (35 degrees C) conditions.

During this evaluation the following takeoff technique was used. The helicopter was stabilized at a 2-foot skid height hover.

Takeoff power was applied and a level acceleration at the 2-foot skid height was made. Approximately 2 to 5 knots before the desired climb-out airspeed was reached, rotation to climb-out attitude was initiated. The climb-out was then accomplished at constant airspeed. Throughout the takeoff, rotor speed was held constant and power was maintained at the maximum takeoff power attainable.

A second takeoff technique, which consisted of a simultaneous climb and acceleration, was evaluated briefly from a qualitative standpoint. Because of limited time available, data were not obtained utilizing the second technique.

During the takeoff performance testing, it was noted that at the 2-foot skid height, as forward flight was initiated (5-10 KTAS), a fairly large decrease in turbine outlet temperature occurred. This decrease required an increase in collective pitch to restore the power turbine outlet temperature to the takeoff limit. This resulted in an increase in torque available during the takeoff run. The cause of this variation in turbine outlet temperature at low speeds is not presently known. The power change was too large to be accounted for by ram effect at the low speeds where the effect was noticed. No unusual inlet temperature rise was recorded during hovering flight at low skid heights. A possible cause might be an increase in exhaust back pressure during hover close to the ground (2-foot skid height) due to the close proximity of the ground to the exhaust exit. If the OH-6A helicopter is considered for production, additional testing should be accomplished to define this phenomenon because the power available in hovering flight at low skid heights is probably reduced at least 10 Shaft Horsepower (SHP).

Some of the summary takeoff performance presented in this report represents extrapolation of available data and is considered to be approximate. The tests were conducted at the higher elevations where power was limited by turbine outlet temperature. At the lower altitudes (below critical altitude for the engine) power was limited by torque. Under these conditions, maximum torque was available throughout the takeoff. As pointed out in the previous paragraph, above critical altitude, an increase in torque was available after initiating forward flight. The extent of this effect should be determined during further tests.

b. Flight Characteristics

During the transition from hover to forward flight during a level acceleration, longitudinal cyclic-collective pitch control coupling was encountered. This coupling was such that when the collective pitch was increased, forward cyclic control was required. Since the helicopter was rotated slightly prior to the application of takeoff power in order to maintain the 2-foot skid height, longitudinal cyclic-collective coupling was not objectionable during takeoff.

Translational lift was characterized by a momentary increase in vibration level (4 per revolution (per rev)) and a tendency for the helicopter to pitch nose-up. The vibration was not excessive and the pitch-up was easily controlled by the use of forward cyclic control. Practice was required to allow anticipation of this nose-up pitching so as to maintain a level acceleration. At conditions in which relatively large values of excess power were available, the helicopter pitch attitude was uncomfortably nose-down during acceleration. This occurred after translational lift had been achieved and as the level acceleration was continued. Takeoffs under these conditions would normally not be made since the large amount of excess power would allow the helicopter to climb out vertically.

Rotation to the pitch attitude corresponding to the desired climb-out airspeed was easily accomplished by the use of aft cyclic control. Rotation was initiated approximately 2 to 5 knots slower than the desired climb-out airspeed.

c. Power Management

One significant factor affecting power management during takeoff was the configuration of the power turbine speed (N_2) beep control switch on the collective pitch stick. The location and design of this control made it impractical to adjust the N_2 rpm once a takeoff was started. A rotor speed 2-3 rpm higher than the desired 469 (100 percent N_2) was selected while hovering to compensate for the steady-state droop which occurred when takeoff power was applied. When operating in the higher power ranges during takeoffs, the engine response rates and acceleration rates were adequate. Large rotor speed variations caused by transient droop were not encountered.

2.2 HOVER

2.2.1 OBJECTIVE

Hovering tests were conducted to determine hovering performance of the OH-6A helicopter. Data from these tests were used to check compliance with the Military Characteristics and determine if the hovering contractual guarantees were met.

2.2.2 METHOD

Hovering performance data were obtained using the free flight method, with data being recorded at various pre-selected skid heights both in-ground effect (IGE) and out-of-ground effect (OGE). A weighted cord of the desired length was used in conjunction with hand signals from a ground observer to obtain precisely the desired skid heights. Data were recorded at stabilized skid heights of 2,5 and 10 feet (IGE) and 50 feet (OGE) in zero wind conditions. These tests were performed at gross weights up to the maximum overload at a mid C.G. location (Station 100.5) and rotor speeds of 464,469 and 482 rpm. Testing was accomplished at density altitudes from sea level to 10,000 feet.

2.2.3 RESULTS

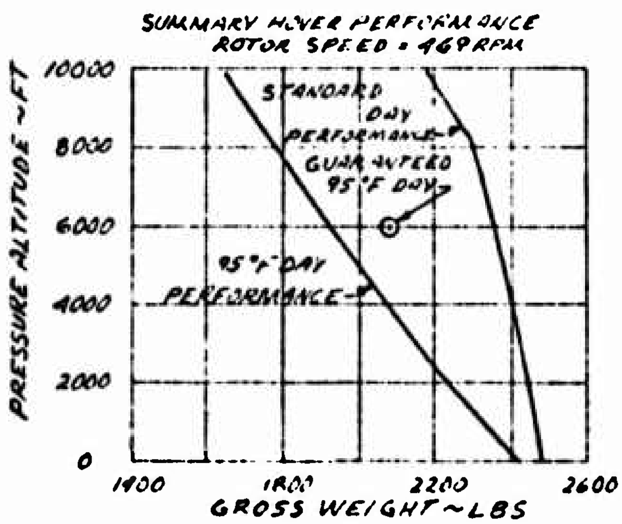
Test results are presented graphically in Figures No. 11 through 19 and summarized in Figures No. 9 and 10, Section 3, Appendix I.

2.2.4 ANALYSIS

a. Guarantee

The OH-6A did not meet the 6000 foot altitude 95 degree F (35 degrees C) hovering performance guarantee. The following Figure B, Summary Hovering Performance, is a plot of the variation of OGE hover ceiling with gross weight for both standard day conditions and the 95 degree F condition:

FIGURE B



From this plot (Figure B), it can be seen that the 95 degree F hover ceiling at normal gross weight was determined to be approximately 3800 feet, an altitude substantially lower (37 percent) than the contract guaranteed ceiling of 6000 feet. This Figure B also shows the lack of OGE hover capability at the overload gross weight condition. The maximum gross weight at which OGE hover can be achieved on a standard day at sea level is 2490 pounds. The Military Characteristics, which require the same hovering capability as guaranteed by the manufacturer, were not met. This hovering performance was determined using the power available as contained in the Model Specification No. 580-A for the T63-A engine.

b. Performance

The summary hovering performance presented in Figures No. 9 and 10, Section 3, Appendix I, was based on the power available for the T63-A-5 as defined by engine Model Specification No. 580-E (See Section 3, Appendix II). An examination of these summary plots reveals that less performance is available using the 580-E Model Specification than is provided by the 580-A Model Specification for conditions above the critical altitude of the engine.

One significant characteristic that is apparent in all the Summary Hovering Performance figures included in this report is the rapid deterioration of hovering performance with increasing ambient temperatures. This characteristic is due primarily to the loss in power with increasing ambient temperatures and not to the helicopter design.

c. Flight Characteristics

No unusual flight characteristics were observed during the hovering tests conducted. Generally, more cyclic stick control motion was required during a stabilized hover close to the ground (2-foot skid height) than was required when hovering farther away from the ground (5 feet skid height and above).

2.3 CLIMBS

2.3.1 OBJECTIVE

Climb tests were conducted to determine the performance during climbing flight, the service ceiling and the best climb airspeed.

2.3.2 METHOD

Continuous climb performance tests were conducted from sea level to service ceiling at two gross weights using maximum continuous power. Takeoff power climbs, for five minutes, were also conducted from sea level at the same gross weights. During the climb, power was maintained at limit engine torque until an altitude was reached at which the corresponding limit turbine outlet temperature was obtained (critical altitude). As the climb continued above this altitude, power was adjusted to maintain the limit turbine outlet temperature.

Sawtooth climbs (a series of short climbs through a specific altitude) were conducted at various airspeeds. These tests were accomplished at altitudes of 5000 and 10,000 feet using maximum continuous and takeoff power settings. The results of these tests were compared with the airspeed for minimum power required in level flight to determine the best climb speed schedule.

Both the continuous and sawtooth climbs were conducted at gross weights of 2085 pounds (design) and 2700 pounds (overload) at a mid C.G. location (Station 100) and a rotor speed of 469 rpm.

In addition, sawtooth climbs at various power settings at design gross weight and climbs at various gross weights using maximum continuous power were conducted to obtain the power and gross weight correction factors.

2.3.3 RESULTS

Test results are presented graphically in Figures No. 20 through 24, Section 3, Appendix I.

2.3.4 ANALYSIS

The climb performance of the OH-6A was normal for a helicopter with its gross weight and installed power. The rate of climb on a sea level standard day at 2085 pounds (design gross weight) was 1470 feet per minute using maximum continuous power. Under the same conditions, at 2700 pounds (overload gross weight), the rate of climb was approximately 900 feet per minute. The rate of climb at 2085 pounds increased slightly as altitude was increased from sea level up to the critical altitude for the engine. At 2700 pounds, the sea level rate of climb value decreased as altitude was increased to critical altitude. The critical altitude for maximum continuous power was approximately 9000 feet, depending on the climb speed. The service ceiling at 2085 pounds and 2700 pounds climb start gross weight are presented in Figures 20 and 21, Section 3, Appendix I at 19,000 and 11,600 feet respectively, using maximum continuous power.

At takeoff power, the rate of climb on a standard day was approximately 1980 feet per minute at 2085 pounds gross weight. At a gross weight of 2700 pounds a sea level rate of climb of 1250 feet per minute was obtained. A similar change in rate of climb with increasing altitude was experienced using takeoff power as was explained above for maximum continuous power. The critical altitude for takeoff power was approximately 7500 feet. Service ceilings using takeoff power were not determined during this evaluation due to the 5 minute power rating.

The most significant flight characteristic noted during the climb tests was a region of random dynamic instability. This occurred at the best climb speed at high rates of climb using zero sideslip. This phenomenon is described in detail in Part I of this report. Briefly, random pitch and yaw oscillations occurred with controls fixed. The magnitude of the oscillations required constant and relatively large control inputs in order to maintain the desired climb airspeeds. In order to improve the handling qualities, it is desirable to climb at some airspeed other than that for best rate of climb. This would result in a decrease in climb performance.

A second characteristic observed during the climb tests was the lack of adequate forward stick force trim authority. During climbs at maximum continuous power at normal gross weight, full forward stick force trim was obtained at approximately 12,000 feet. As the climb continued above this altitude, forward stick force was required to maintain the desired airspeed. At 17,000 feet approximately 8-10 pounds of forward stick force was required.

No significant power management problems were encountered during the climb tests. Climbs were entered from level flight, and engine response and acceleration characteristics were satisfactory during climb entries. Difficulty was encountered in achieving a stabilized torque indication during climb entries. This, however, was due to lag in the torque indication system rather than a lack of engine response. During the portion of the climb below critical altitude, collective control was gradually increased to maintain the desired torque value. Occasional small adjustments in N_2 rpm were required during this portion of the climb since some droop in rotor rpm did occur. Once the critical altitude was reached and the turbine outlet temperature limit obtained, collective control remained essentially fixed for the remainder of the climb.

2.4 LEVEL FLIGHT

2.4.1 OBJECTIVE

Tests were conducted in level flight to determine the range, endurance, speed and power required by the OH-6A helicopter. Tests with the XM-7 and XM-8 weapon systems installed were also conducted to determine their effect on helicopter performance. Data from these tests were used to evaluate compliance with the Military Characteristics and to determine if the contractual guarantees were met.

2.4.2 METHOD

Speed power tests were conducted at various conditions of altitude, gross weight and rotor speed in both the unarmed (clean) and armed configurations. Each speed power was flown at a constant value of gross weight divided by density (W/ρ) so that a comparative analysis could be made. This involved increasing altitude as fuel was consumed. During the tests, data were recorded in stabilized level flight at various airspeeds throughout the allowable speed range at approximately 10 knot increments so as to define adequately the particular power required curve. In addition to basic power parameters, fuel flow data were also recorded.

Tests in the clean configuration were conducted at density altitudes of approximately sea level, 5000, 10,000 and 15,000 feet. Gross weights of approximately 1890, 2100 and 2700 pounds were used at a mid C.G. location (Station 100.3). Rotor speeds of 463, 469 and 482 rpm were flown. Two additional tests were conducted at an altitude of 5000 feet, a gross weight of 2100 pounds, with a rotor speed of 469 rpm. One of these tests was at a forward C.G. location (Station 96) and the other at an aft C.G. location (Station 103).

Tests in the armed configurations were conducted at density altitudes of approximately 2000 and 5000 feet, a gross weight of 2090 pounds, at a mid C.G. location (Station 100.3) and with a rotor speed of 469 rpm.

2.4.3 RESULTS

Test results are presented graphically in Figures No. 29 through 45 and summarized non-dimensionally in Figures No. 26 through 28, Section 3, Appendix I.

2.4.4 ANALYSIS

a. Guarantee

The OH-6A complied with the maximum airspeed guarantee of 123 KTAS ± 10 percent (135.3 to 110.7). Figure C on page 14 shows that the maximum airspeed obtained at design gross weight, sea level and normal rated power was 120 KTAS. This airspeed was within the ± 10 percent tolerance of the guaranteed 123 knots.

The OH-6A did not meet the guaranteed endurance performance of 3.4 hours ± 10 percent (3.74 to 3.06 hours). The maximum endurance was found to be 3.01 hours based on performance calculation (See Figures No. 26 through 28, Section 3, Appendix I), and the contractor's guaranteed endurance mission profile. Figure D, page 14, shows the variation of specific range with gross weight at the airspeed for the guaranteed endurance (100 knots).

FIGURE C

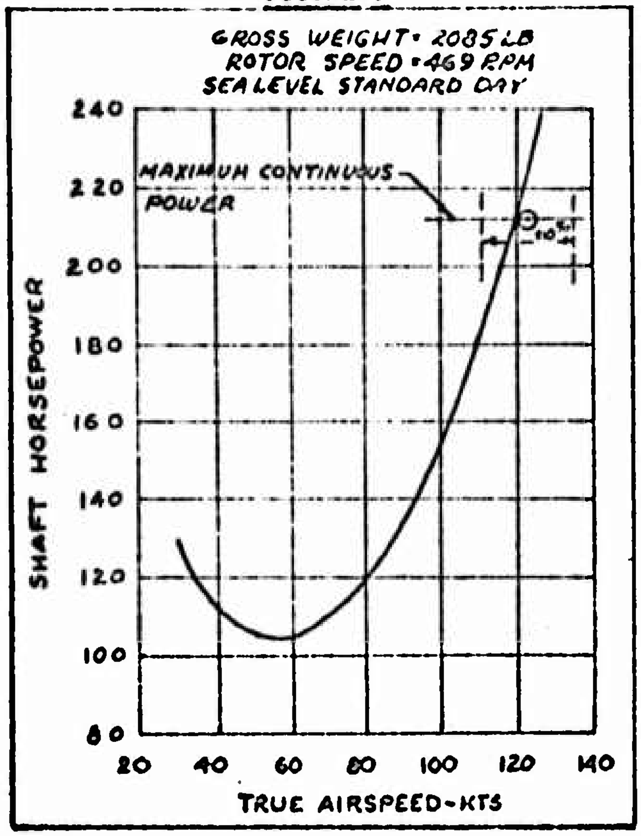
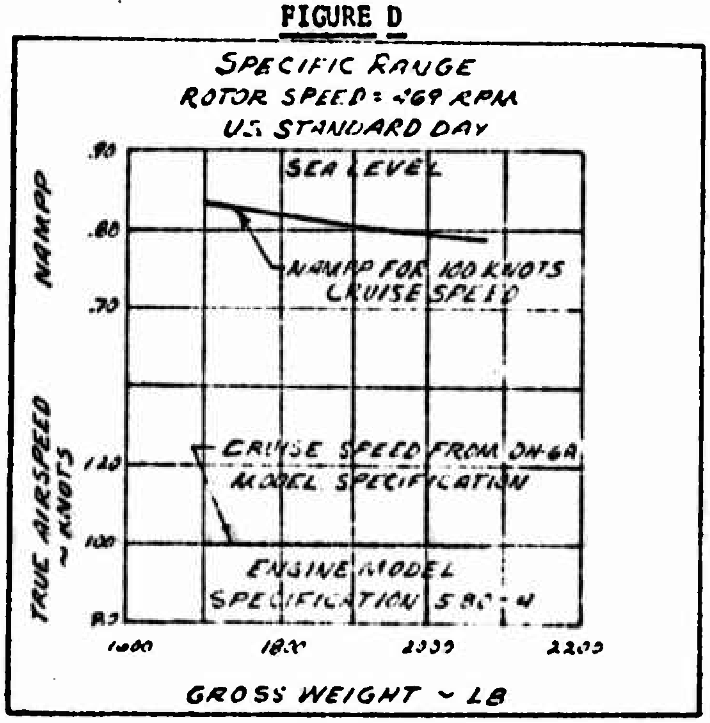


FIGURE D



The mission profile consisted of:

(1) Warmup and takeoff: 3 minutes at normal rated power at sea level.

(2) Cruise at 100 KTAS at sea level.

(3) Reserve: None.

(4) Useful load at normal gross weight for the endurance mission:

Pilot (1)	-	200 lb
Observer (1) and cargo	-	400 lb
Fuel (JP-4)(as necessary for endurance problem)	-	59 gal max.
Oil (as necessary for above fuel)		

The following table shows the comparison of the Military Characteristics performance requirements to the performance of the OH-6A:

	Military Characteristics	OH-6A
Payload (Useful load less 200 lb pilot oil and 3.0 hrs fuel)	- 400 lb	400 lb
Cruise Speed	- 110 kt	110 kt
Endurance at 85% of Cruise Power	- 3 hr	3.01 hr

b. Performance

There was sufficient power available, using the takeoff power rating, to reach Vne at gross weights from design to overload up to altitudes of 10,000 feet (See Part I, Section 2). At airspeeds near Vne, the OH-6A demonstrated undesirable pitch-up characteristics. In general, the severity of the pitch-up varied with gross weight and altitude but was noted to some degree at all conditions tested.

At gross weights of 2100 pounds or less, the Vne exceeded the recommended cruise speed. At gross weights in excess of 2100 pounds, the Vne restriction for 2700 pounds was applicable. This limited the aircraft to an airspeed less than that which would produce the optimum range. The maximum range at gross weights of 2200 and 2300 pounds is reduced substantially since the difference between best cruise speed and Vne was the largest. A decrease of 20 KTAS from the best cruise speed (106 KTAS) at 2200 pounds and at 8740 feet is shown in Figure No. 37, Section 3, Appendix I.

For a given gross weight, altitude and rpm, the forward and aft C.G. level flight and range performance was similar to the performance at a mid C.G.

The addition of either of the XM-8 or XM-7 armament kits reduced the speed at maximum continuous power less than 5 knots. The increase in flat plate area for the XM-7 was .700 square feet and .950 square feet for the XM-8 kit at the airspeed for maximum continuous power.

Power management during level flight did not present any unusual problems. The droop cam compensator was satisfactory. When stabilized airspeeds were selected at 469 rpm, from approximately 50 KIAS to Vmax, a maximum variation of 3 to 4 rotor rpm (less than 1 percent) was encountered.

2.5 AUTOROTATIONAL DESCENTS

2.5.1 OBJECTIVES

Testing was accomplished to determine optimum rotor speeds and airspeeds for autorotational descents. In addition, data were obtained to allow determination of rates and angles of descent during autorotation.

2.5.2 METHOD

Autorotational descents were conducted at various airspeeds and rotor speed combinations. During the descent, time and altitude were recorded so that rate of descent could be determined.

A series of stabilized descents were conducted at a rotor speed of 469 rpm and at various airspeeds between 30 and 80 knots calibrated airspeed (KCAS). From these tests, the airspeed for minimum rate of descent was determined. At the airspeed for minimum rate of descent, a second series of descents were conducted at rotor speeds from 440 to 480 rpm.

The autorotational descents were conducted at density altitudes from approximately 10,000 to 1,900 feet. Various gross weights from 1800 to 2580 pounds were flown at a mid C.G. location.

2.5.3 RESULTS

Test results are presented graphically in Figures No. 46 through 48, Section 3, Appendix I.

2.5.4 ANALYSIS

The airspeed for minimum rate of descent was determined to be 53 KCAS at a rotor speed of 469 rpm. At this airspeed the rate of descent was 1575 feet per minute. This rate of descent is normal for a helicopter of this size and gross weight. The rate of descent was found to be independent of altitude or gross weight (See Figure No. 46, Section 3, Appendix I).

The airspeed for minimum angle of descent was determined to be 71 KCAS at an altitude of 5000 feet at a rotor speed of 469 rpm. At this airspeed, approximately .73 nautical air miles could be traveled per 1000 feet of descent.

The rotor speed for minimum rate of descent was not determined during this test program. True autorotation, during which no power is delivered to the rotor, could not be achieved at rotor speeds less than 440 rpm since this corresponded to the power turbine speed which resulted at the gas producer idle speed. Because of this, autorotational descent data were not obtained at lower rotor speeds. The results obtained indicate the minimum rate of descent would occur at a rotor speed below 440 rpm.

Autorotational entries posed no unusual problems in the OH-6A. No significant trim change, except for the normal directional trim change, accompanied closing the throttle. The initial rpm decay rate while high, was not objectionable. The large range of allowable rotor speed (down to 380 rpm) permits a 2-second delay between closing the throttle and reducing collective pitch throughout most of the level flight envelope, assuming that the initial rotor speed is high. The largest trim change noted during the entry occurred in conjunction with lowering of collective pitch. As collective pitch was lowered, a nose-down pitching occurred. The pitch rate was directly proportional to the rate at which collective was lowered as well as the initial collective setting. This nose-down pitching was easily controlled by the simultaneous application of aft cyclic control.

In stabilized autorotation, the cyclic stick position was uncomfortably aft and slightly to the right of center. At higher gross weights, full down collective pitch could result in overspeeding the rotor and careful pilot technique was required. Constant attention to maintain stabilized conditions was required since very small variations in airspeed resulted in relatively large variations in rotor speed.

Power recoveries exhibited some unusual characteristics primarily because of the engine response characteristics and the power turbine override control switch incorporated in the throttle system. This override control beeped the power turbine speed to "full increase" whenever the throttle was closed to the ground idle position. Thus, when the throttle was opened during a power recovery, power turbine speed, and hence rotor speed, would increase to approximately 103 percent. Generally some overshoot occurred that resulted in momentary peaks as high as 108 percent, a value high enough to be disconcerting to the pilot. In addition, the rate at which power was developed varied, with the largest rate of increase occurring at near maximum rpm. This rapid increase in torque was difficult to anticipate and normally resulted in some yaw oscillations of the helicopter as the pilot attempted to correct with antitorque control.

In general, the power recoveries performed fell into two categories. In the first category, the power recoveries were performed at the termination of autorotational descents not in close proximity to the ground and power was increased gradually. In the second category, the power recoveries were made in conjunction with practice autorotational landings, wherein, relatively large power demands were made in a short period of time. For the second case, the characteristics of the T63-A-5 engine installed in the OH-6A were unsatisfactory. The transient power turbine droop was excessive when such large, rapid power demands were made and in some cases, the rotor speed decreased to below the minimum power-on limit.

2.6 AIRSPEED CALIBRATION

2.6.1 OBJECTIVE

The objective of these tests was to determine the airspeed position error for both the standard and test airspeed systems.

2.6.2 METHOD

The airspeed calibration of the standard and test systems was determined by using the ground speed course method. The aircraft was flown over a measured course at various stabilized airspeeds on reciprocal headings. Airspeeds from approximately 20 to 122 knots calibrated airspeed (KCAS), using approximately 10 knot increments, were flown. These tests were conducted at a density altitude of

1670 feet, a gross weight of 2090 pounds, a mid C.G. location (Station 101.3) and a rotor speed of 469 rpm.

2.6.3 RESULTS

Test results are presented graphically in Figures No. 63 and 64, Section 3, Appendix I.

2.6.4 ANALYSIS

The boom airspeed system position error was determined to be + 4.5 knots which was constant over the airspeed range tested. The position error in the aircraft's airspeed system was a maximum of + 7.5 knots at 20 knots indicated airspeed (KIAS) and gradually decreased to +1 knot at airspeeds of 100 KIAS and above. This position error was not considered to be excessive. The aircraft's airspeed system possessed the desirable feature of having a very small position error at high speeds near Vmax. The system became unusable below approximately 15 KIAS. Both the standard and test aircraft systems were not calibrated in climb nor in autorotation since a suitable means of calibration was not available. The boom system utilized a swiveling pitot-static head so as to eliminate errors which might arise due to large sideslips or large variations in angles of attack; hence the level flight calibration was assumed to be correct throughout all flight conditions.

SECTION 3 - APPENDICES

APPENDIX I - TEST DATA

FIGURE NO. 1

SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE

OH-6A USA S/N 62-4212

CLEAN CONFIGURATION

TECHNIQUE: LEVEL ACCELERATION FROM 2FT. HOVER

GROSS WEIGHT - 2085 LB.

ROTOR SPEED = 469 RPM

----- OUT OF GROUND EFFECT HOVER CEILING

TAKE OFF POWER

NOTE:
DERIVED FROM FIGURE 4
AND 16

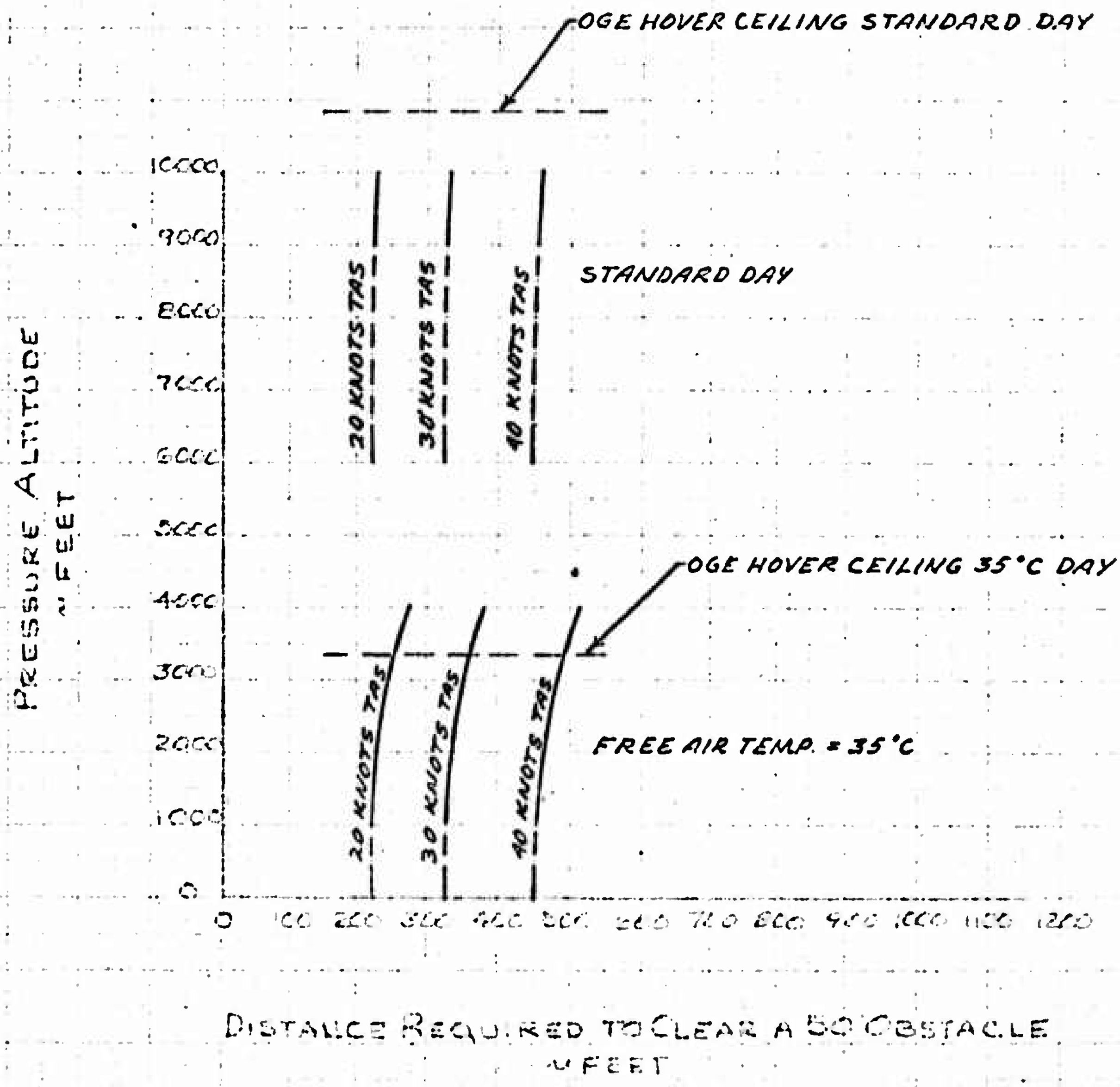


FIGURE NO. 2
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-6A USA S/N 62-4212
CLEAN CONFIGURATION
TECHNIQUE: LEVEL ACCELERATION FROM 2 FT. HOVER
GROSS WEIGHT = 2400 LB
ROTOR SPEED = 469 RPM
– – – – OUT OF GROUND EFFECT HOVER CEILING
TAKE OFF POWER

NOTE:
DERIVED FROM FIGURE 4
AND 16

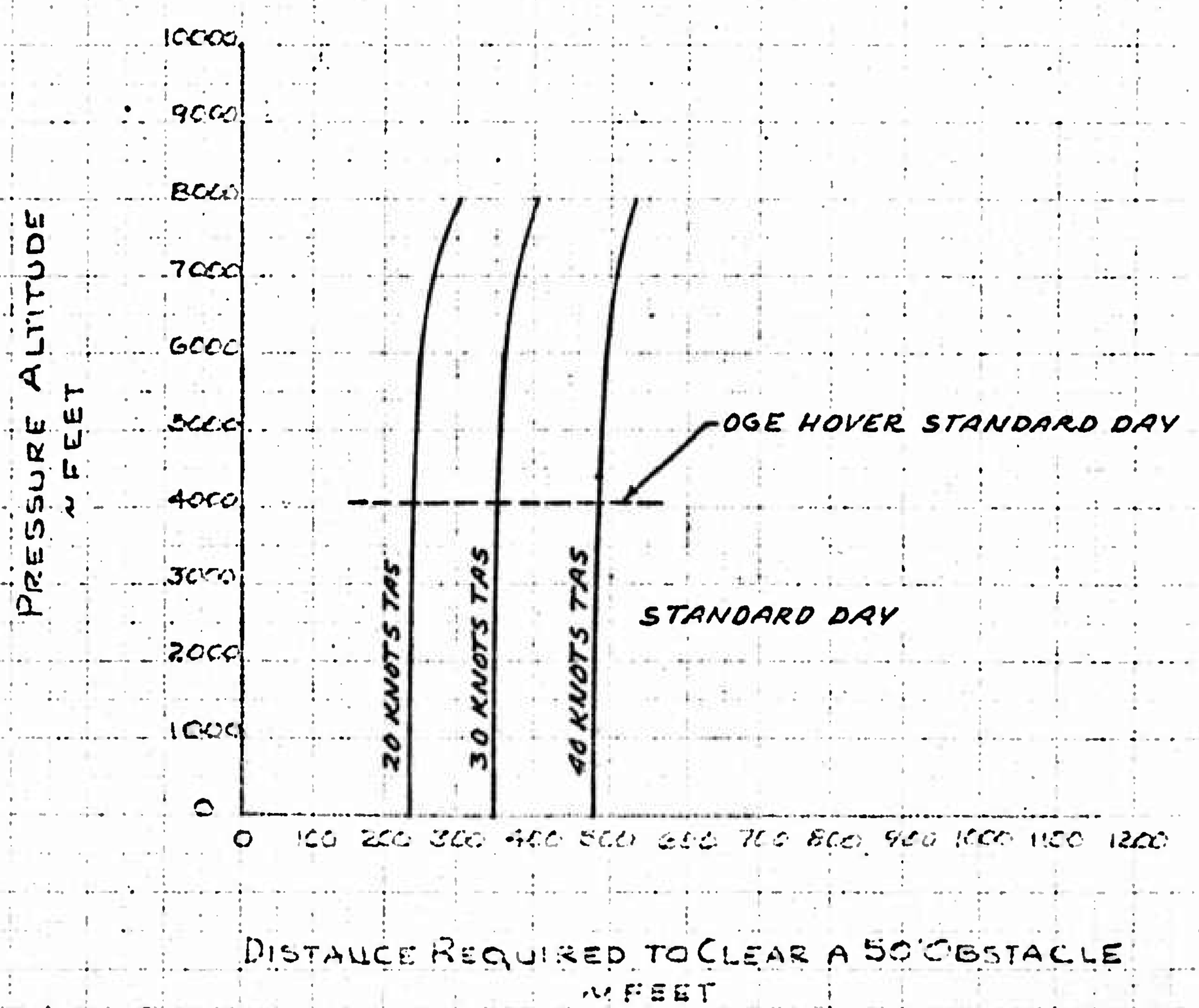


FIGURE NO. 3
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-6A USA S/N 62-4212
CLEAN CONFIGURATION
TECHNIQUE: LEVEL ACCELERATION FROM 2FT. HOVER
GROSS WEIGHT - 2400 LB.
ROTOR SPEED = 469 RPM
TAKE OFF POWER

NOTE:
DERIVED FROM FIGURE 4
AND 16

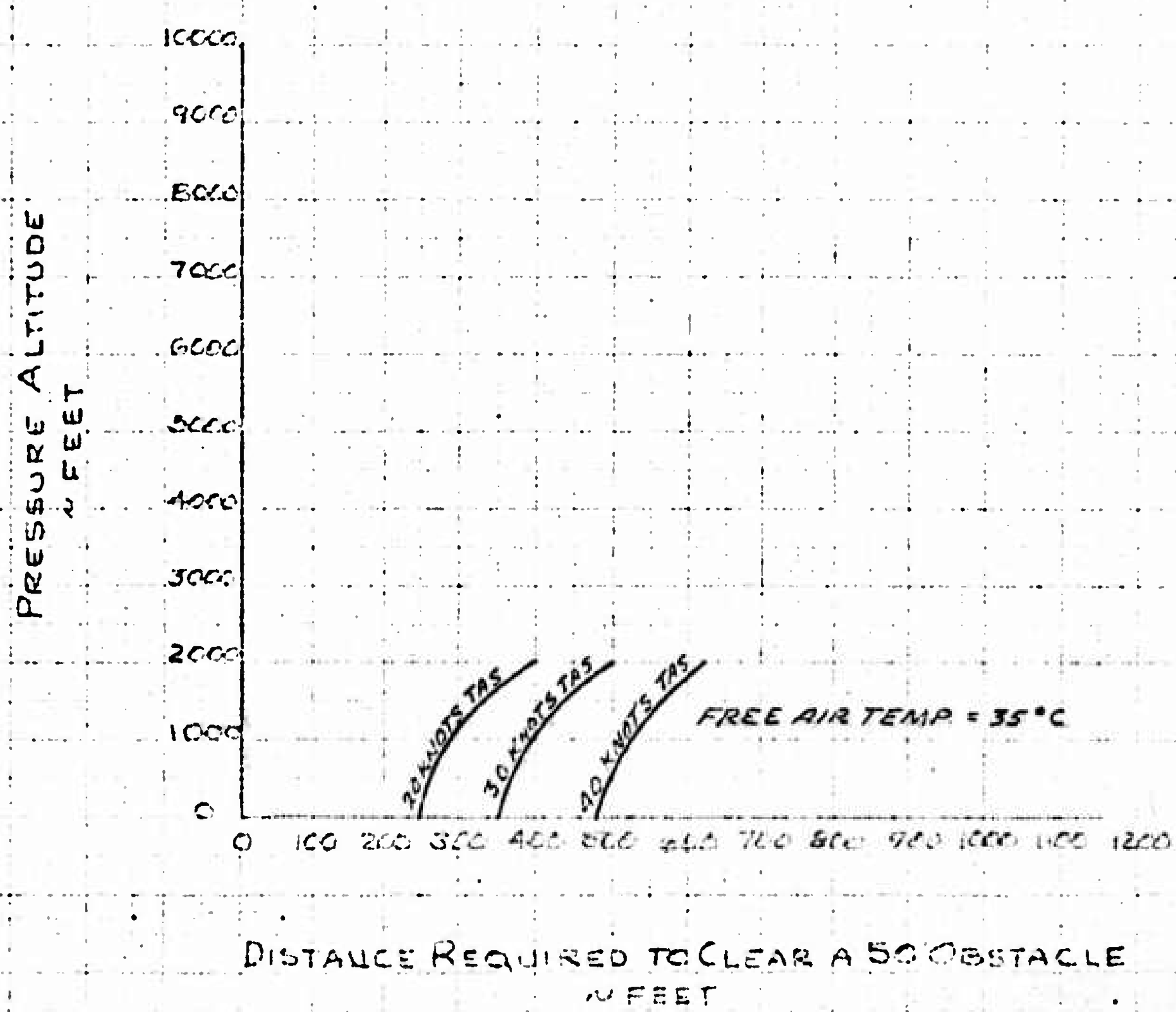


FIGURE NO. 4
VARIATION OF TAKE-OFF DISTANCE WITH CLIMB-OUT AIRSPEED AND ΔC_p
OH-6A USA S/N 62-4212

TECHNIQUE LEVEL ACCELERATION FROM A TWO FOOT HOVER

ROTOR SPEED 469 RPM

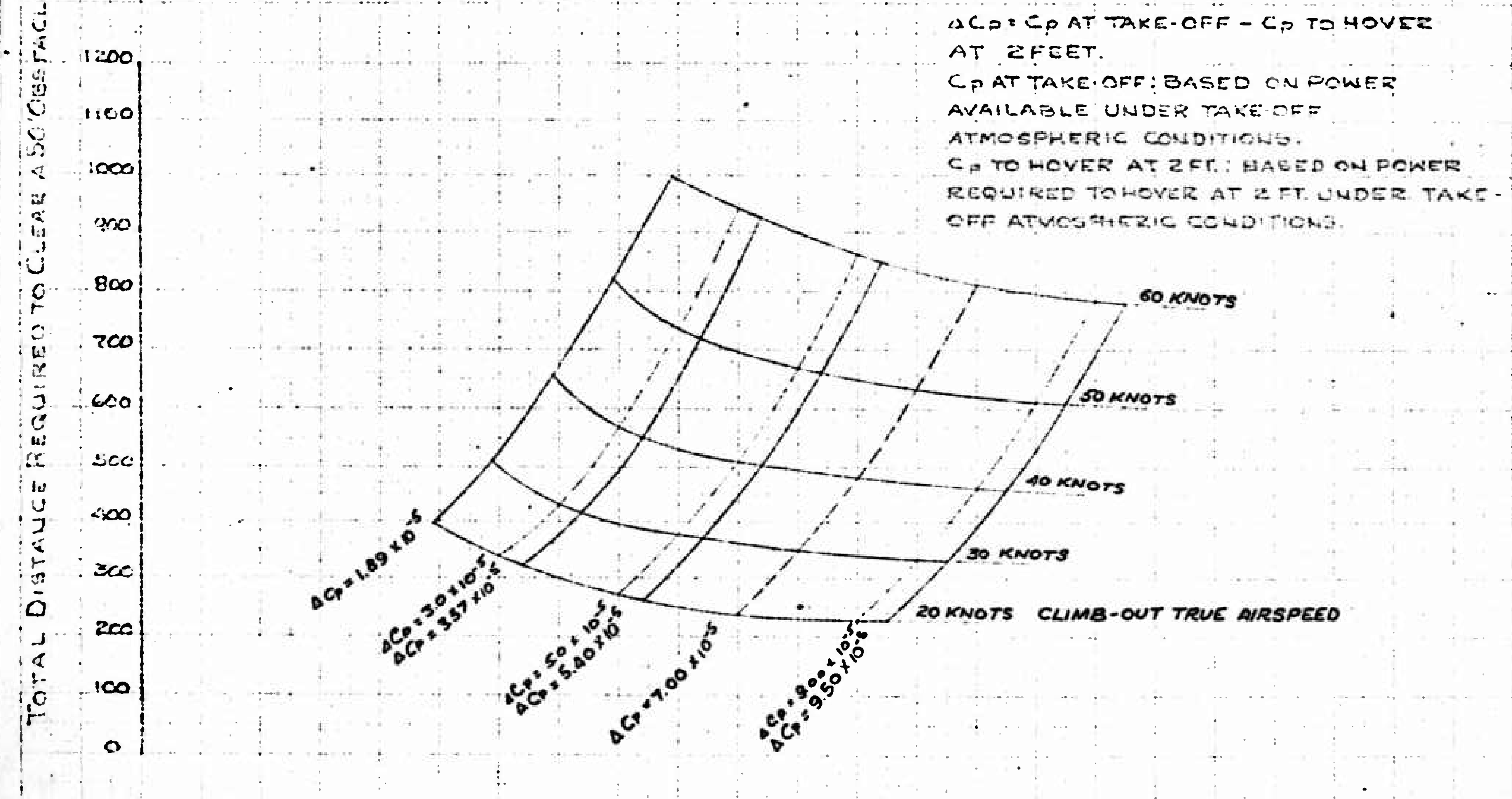


FIGURE No. 5

TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-6A USA S/N 62-4212

ROTOR SPEED	469
GROSS WEIGHT	2050
FREE AIR TEMP.	40°C
WIND	≤ 2 KNOTS
ΔC_P	1.89×10^{-5}
PRESSURE ALT.	9630 FT.
TECHNIQUE	LEVEL ACCELERATION FROM A 2 FT HOVER

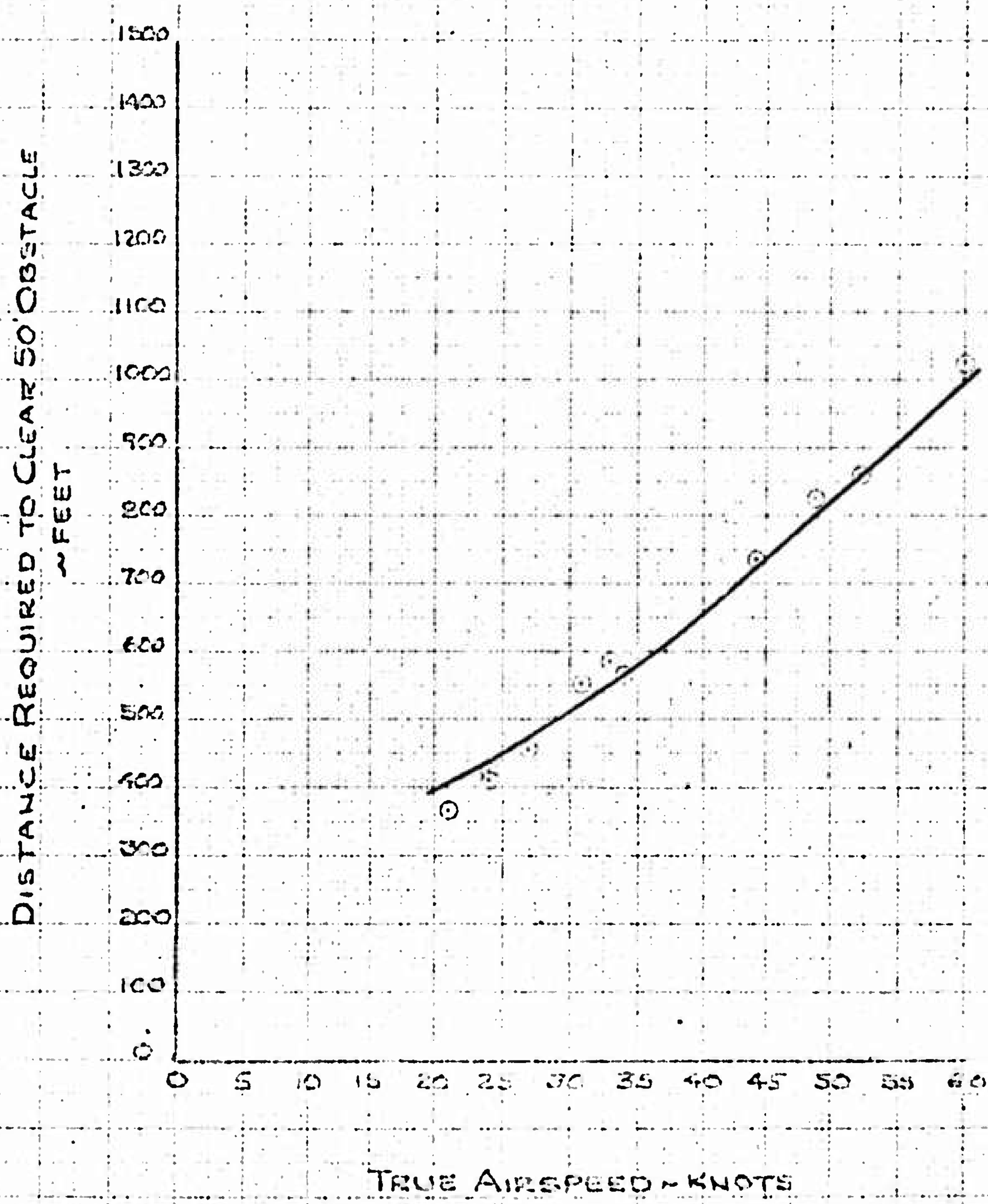


FIGURE NO. 6
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-6A USA S/N 62-4212

ROTOR SPEED 469 RPM
GROSS WEIGHT 2030 LBS
FREE AIR TEMP. 2.5 °C
WIND ≤ 2 KNOTS
ΔC_P 3.37×10^{-5}
PRESSURE ALT. 9650 FT.
TECHNIQUE LEVEL ACCLERATION FROM A 2 FT HOVER

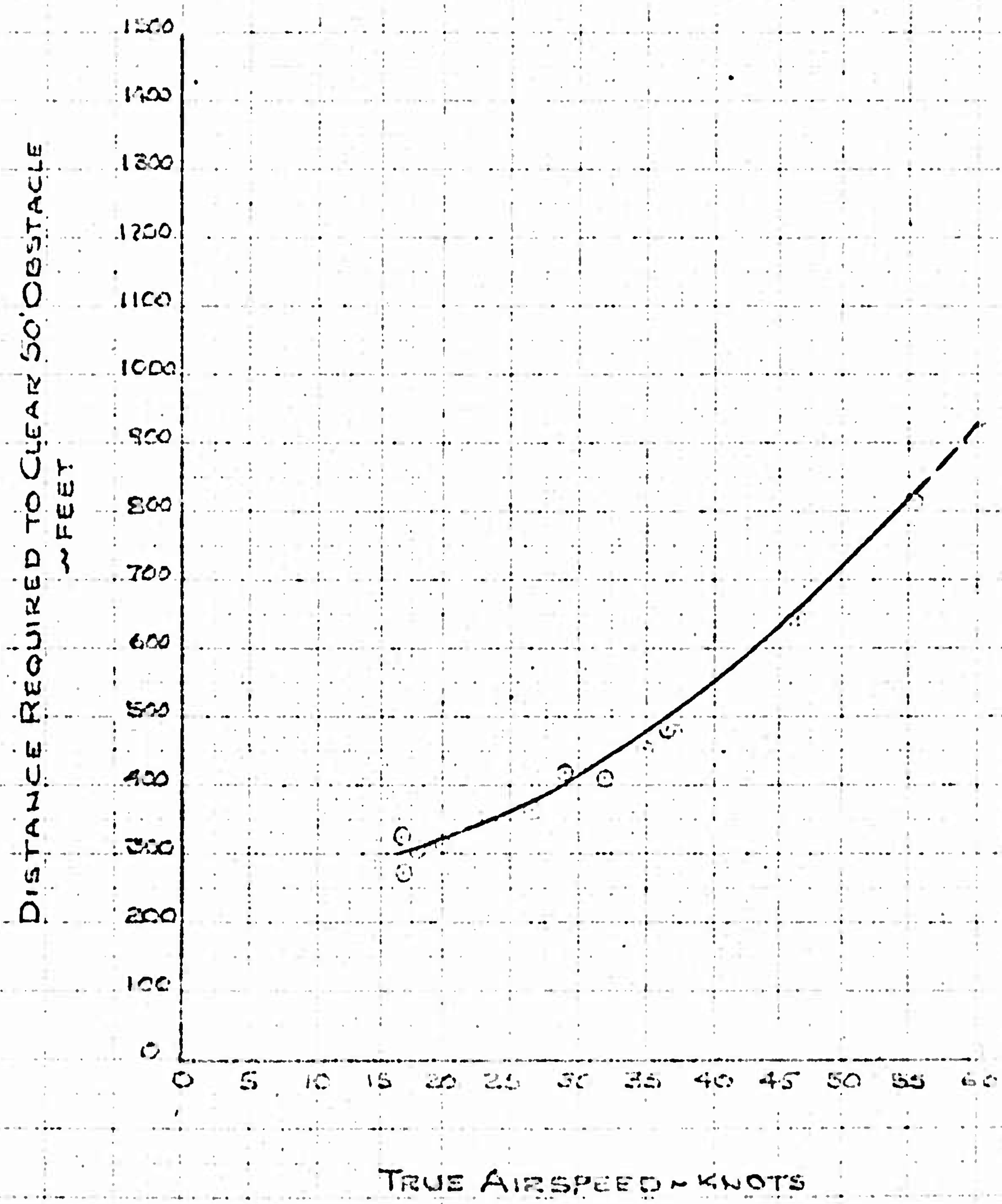


FIGURE NO. 7
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-6A USA S/N 62-4212

ROTOR SPEED	469 RPM
GROSS WEIGHT	1790 LBS
FREE AIR TEMP.	9°C
WIND	≤ 2 KNOTS
ΔC_P	5.40×10^{-5}
PRESSURE ALT.	9450 FT
TECHNIQUE	LEVEL ACCELERATION FROM A 2FT HOVER

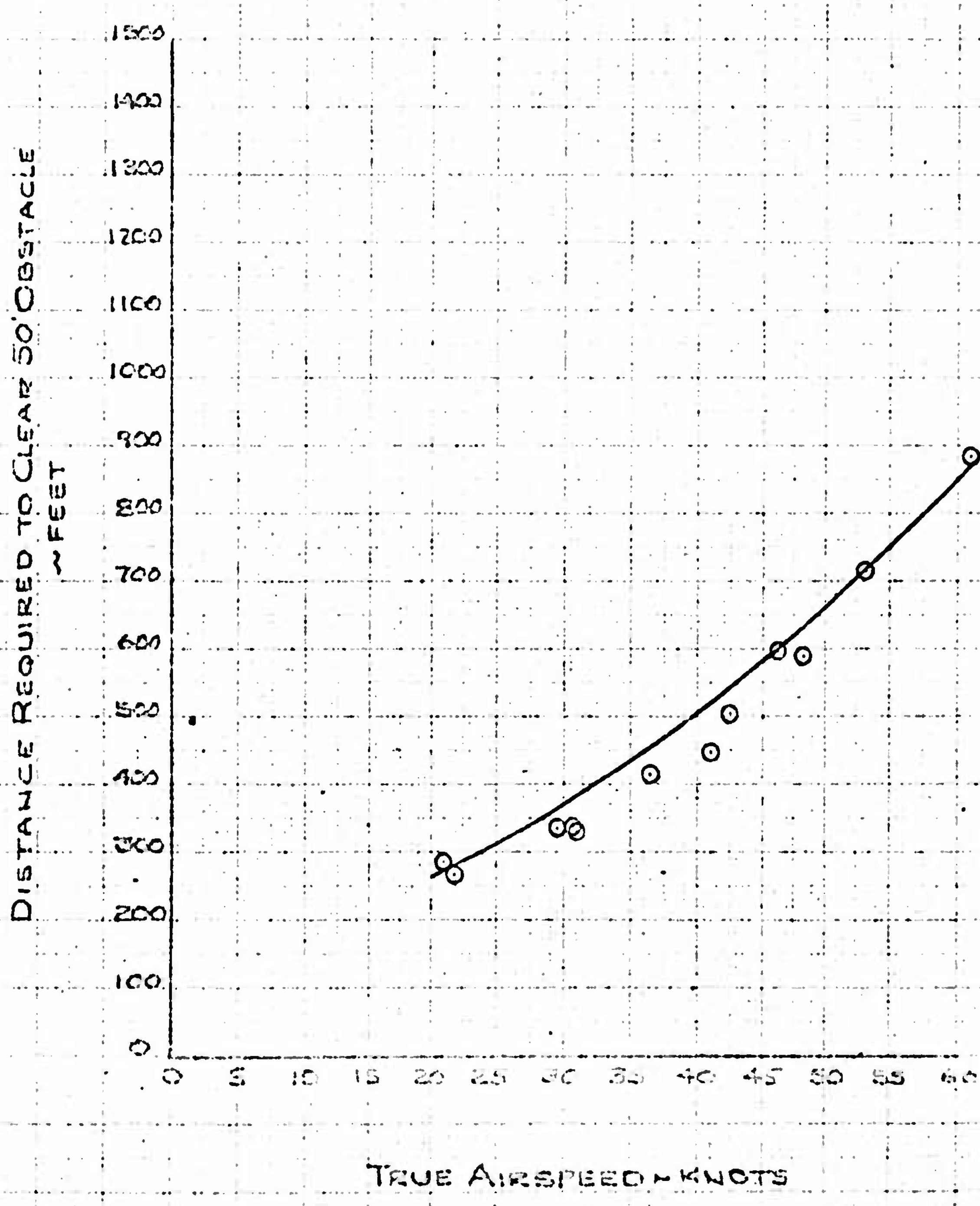


Figure No. 8
Take-Off Distance Required
To Clear A 50' Obstacle
OH-6A USA S/N 62-4212

ROTOR SPEED	469 RPM
GROSS WEIGHT	1900 LBS
FREE AIR TEMP.	25 °C
WIND	≤ 2 KNOTS
ΔC_P	9.50×10^{-5}
PRESSURE ALT.	4200 FT
TECHNIQUE	LEVEL ACCELERATION FROM A 2 FT HOVER

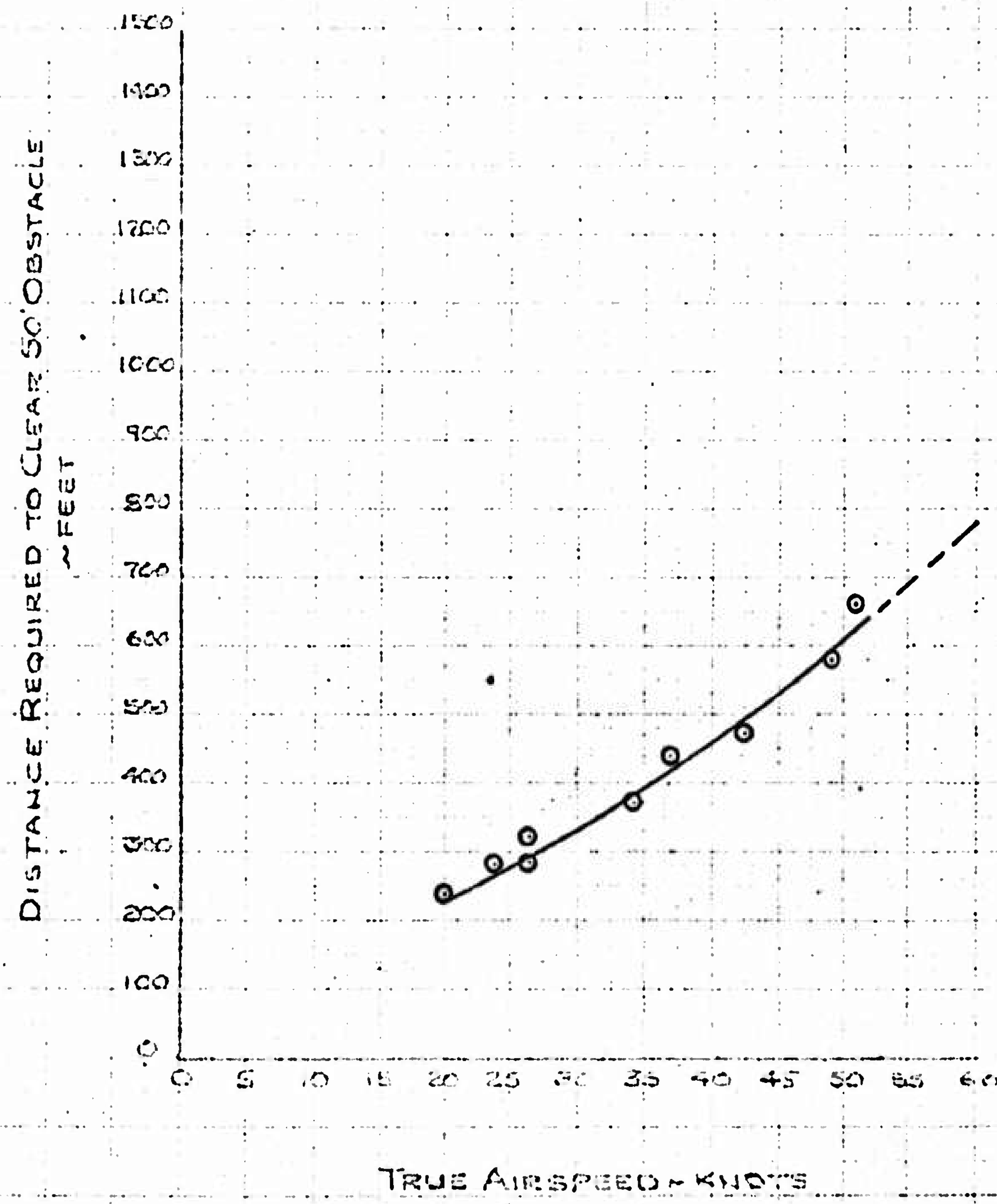


FIGURE No 9
SUMMARY HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
ROTOR SPEED = 469 RPM
TAKE-OFF POWER
CLEAN CONFIGURATION

NOTES
1. BASED ON ENGINE MODEL SPECIFICATION 580-E
2. BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE
3. ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT
4. WIND LESS THAN 2 KNOTS

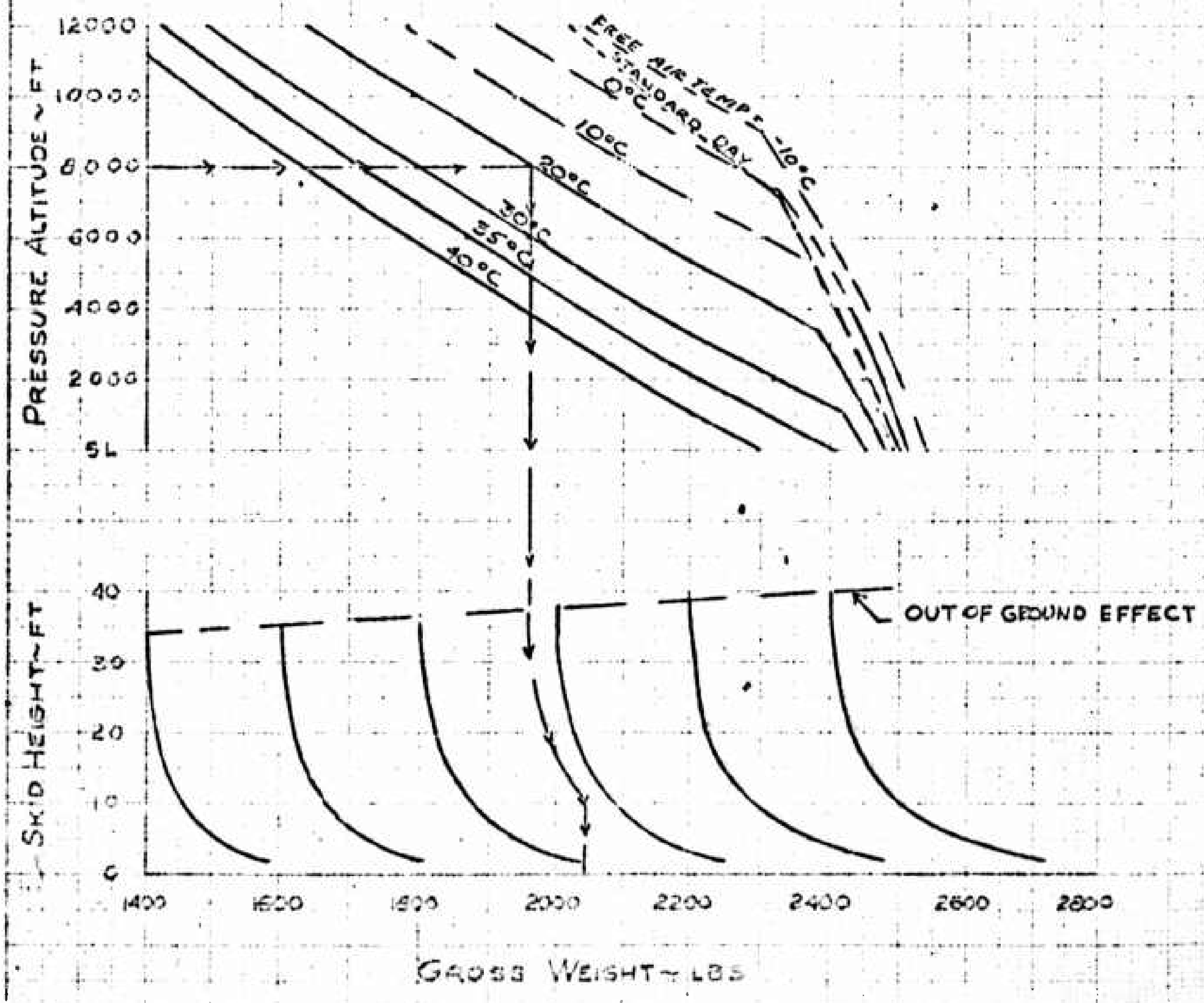


FIGURE No 10
SUMMARY HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
ROTOR SPEED = 469 RPM
MAXIMUM CONTINUOUS POWER
CLEAN CONFIGURATION

NOTES:
1. BASED ON ENGINE MODEL SPECIFICATION 580-E
2. BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE.
3. ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT
4. WIND LESS THAN 2 KNOTS

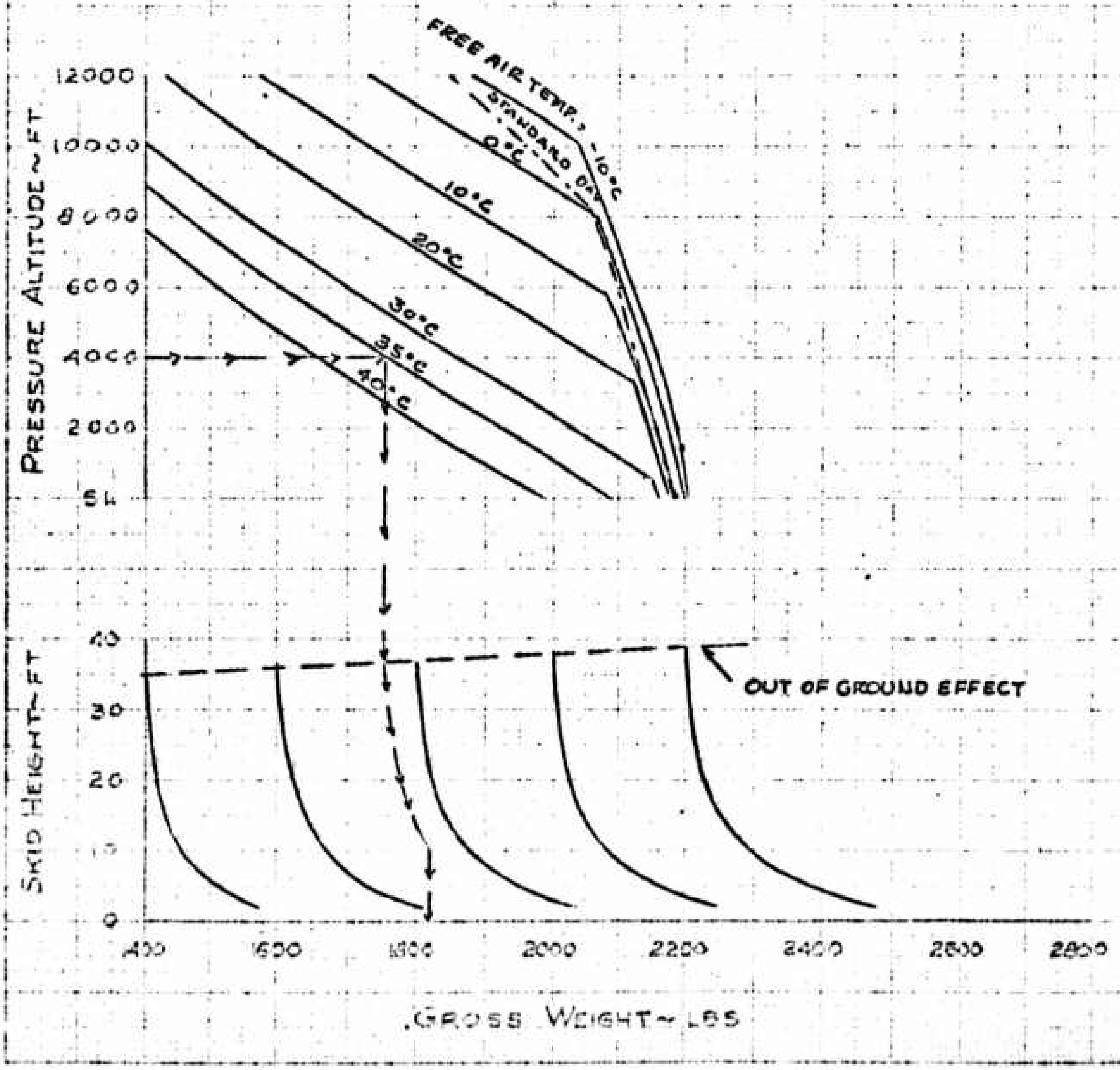


FIGURE No 11

SUMMARY HOVERING PERFORMANCE

OH-6A USA S/N 62-4212

ROTOR SPEED: 469 RPM

TAKE-OFF POWER

OUT OF GROUND EFFECT

CLEAN CONFIGURATION

NOTES

1. BASED ON ENGINE MODEL SPECIFICATION 580-A
2. BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE
3. ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT
4. WIND LESS THAN 2 KNOTS

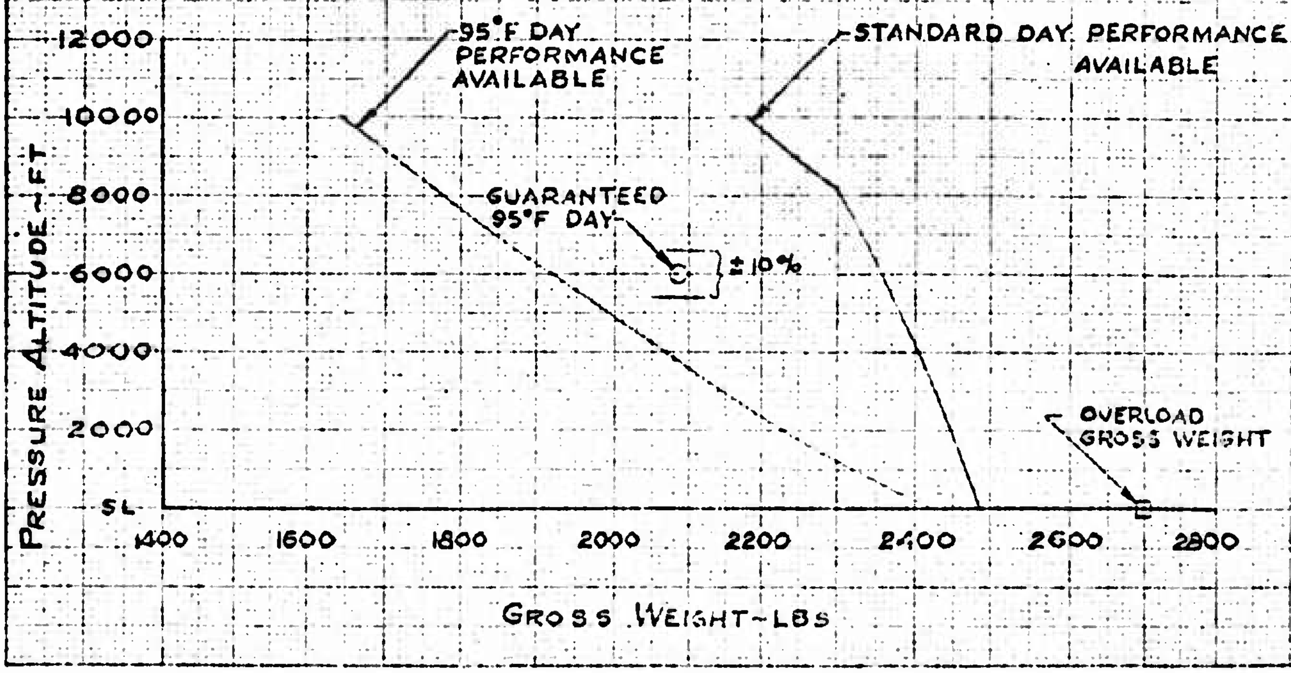


FIGURE No. 12
FUEL FLOW DURING HOVERING FLIGHT
OH-6A USA S/N 62-4212
US STANDARD DAY

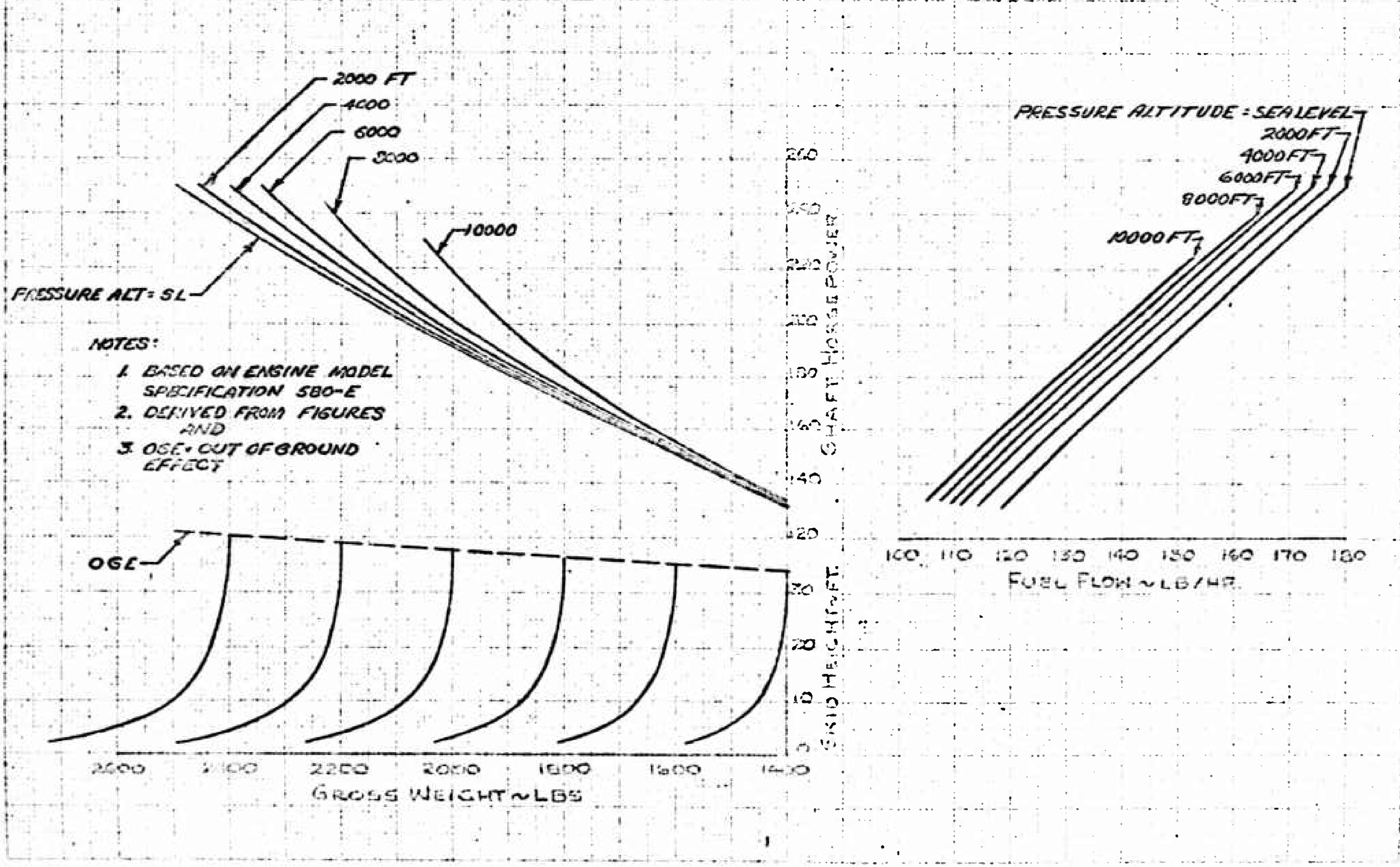


FIGURE NO. 13
NON DIMENSIONAL HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
FREE FLIGHT
CLEAN CONFIGURATION

NOTES:

1. DERIVED FROM FAIRED CURVES ON FIGURE 15
2. VERTICAL DISTANCE FROM BOTTOM OF SKIDS TO CENTER OF ROTOR HUB = 7.66 FT.

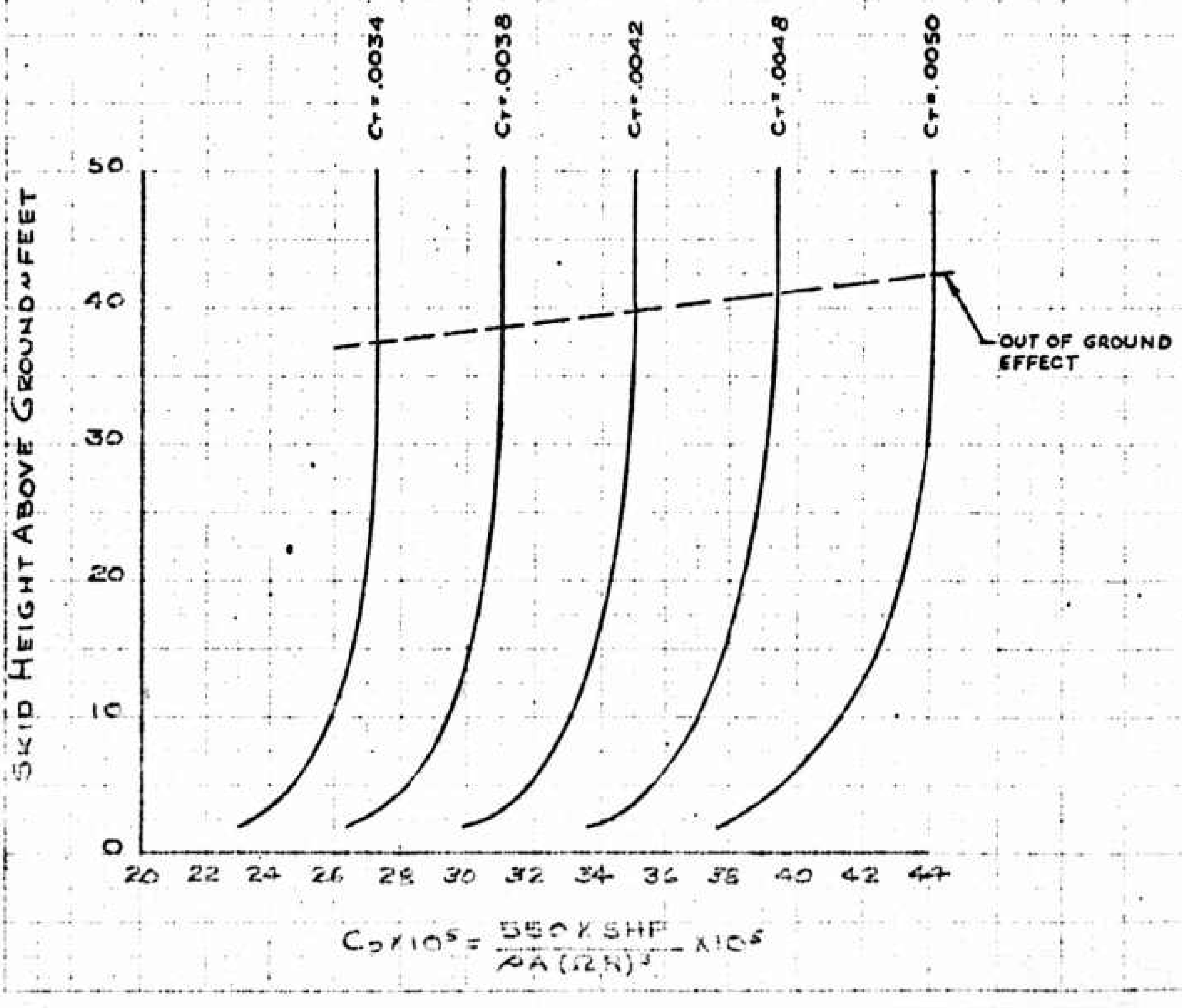


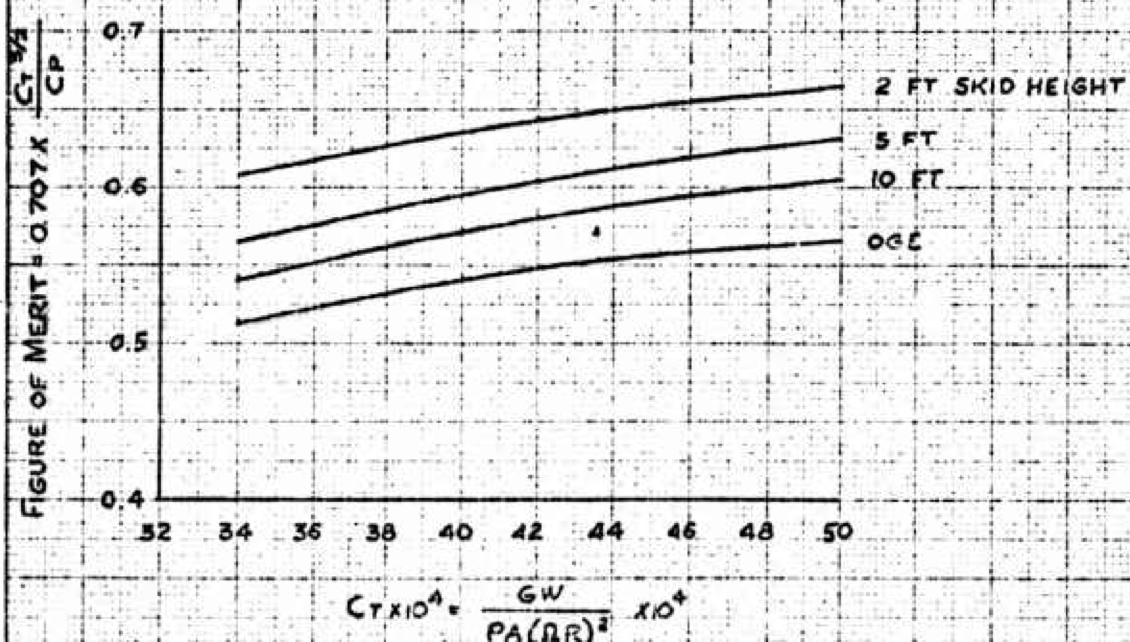
Figure No 14

Non Dimensional Hovering Performance

OH-6A USA S/N 62-4212

NOTES:

1. DERIVED FROM FIGURES 16 THROUGH 19
2. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 7.66 FT.
3. OGE = OUT OF GROUND EFFECT
4. WIND LESS THAN 2 KNOTS



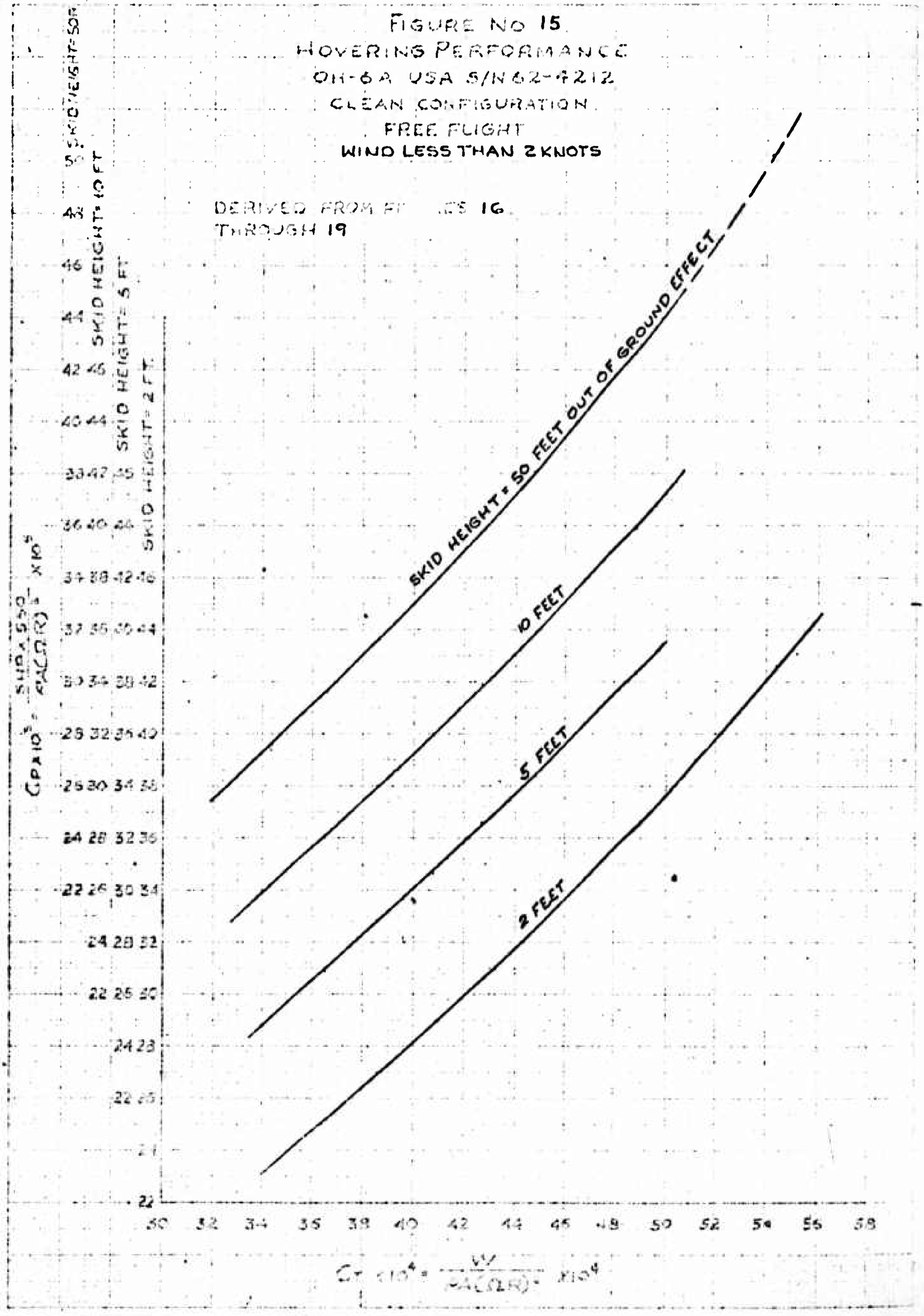


FIGURE NO 15
HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
CLEAN CONFIGURATION
FREE FLIGHT

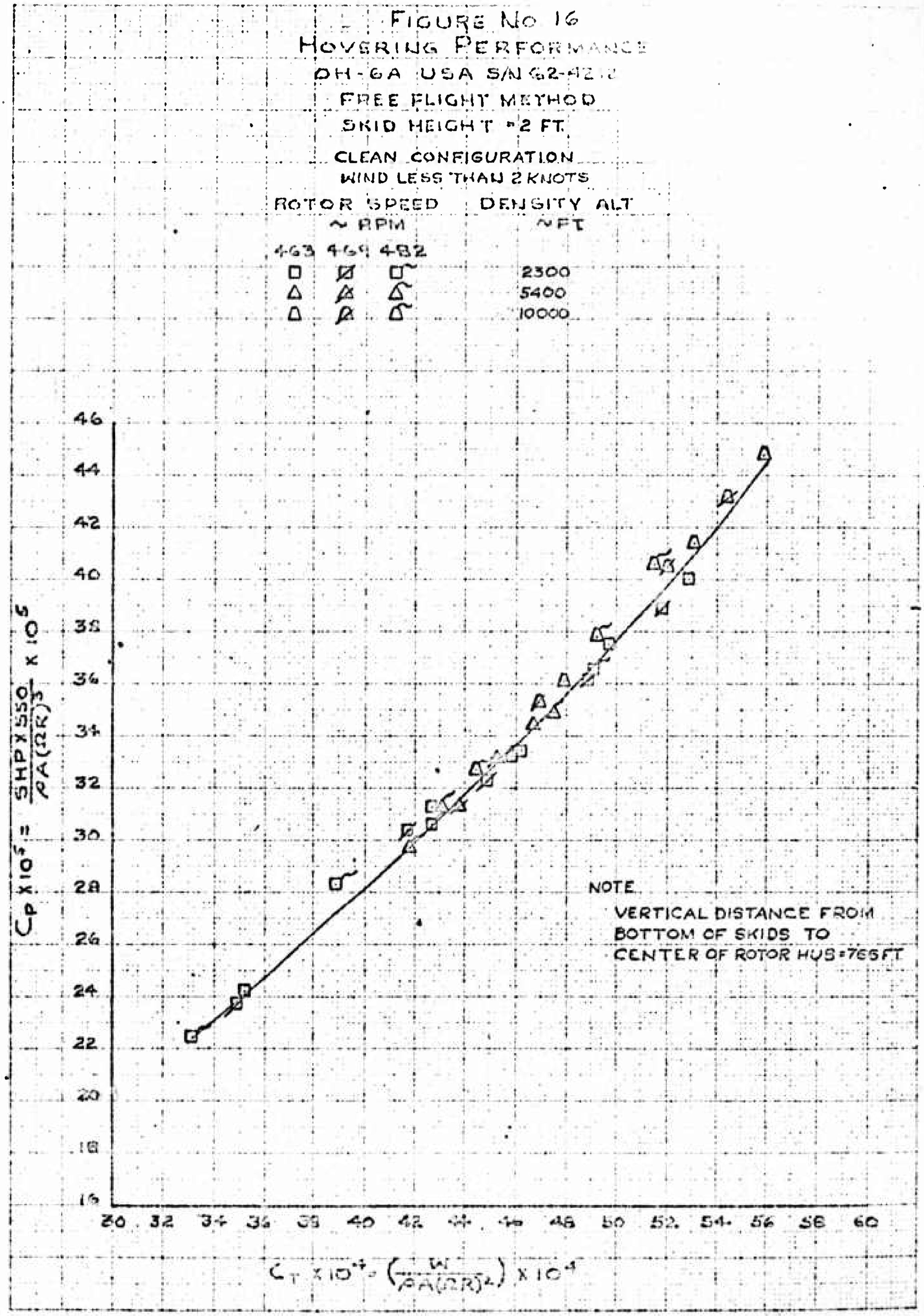


FIGURE NO 16
HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT = 2 FT.
CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS
ROTOR SPEED ~ RPM
DENSITY ALT ~FT
463 469 482
2300
5400
10000
NOTE
VERTICAL DISTANCE FROM BOTTOM OF SKIDS TO CENTER OF ROTOR HUB = 7.65 FT
$C_P \times 10^5 = \frac{SHP \times 550}{\rho A (\Omega R)^3} \times 10^5$
$C_T \times 10^4 = \left(\frac{W}{\rho A (\Omega R)^2}\right) \times 10^4$
46
44
42
40
38
36
34
32
30
28
26
24
22
20
18
16
30
32
34
36
38
40
42
44
46
48
50
52
54
56
58
60

FIGURE NO 17
HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT = 5 FT
CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS

ROTOR SPEED ~ RPM			DENSITY ALT ~ FT
463	469	482	
○	⌀	○̃	600
△	△̸	△̃	5400
◇	◇̸	◇̃	7100
△	△̸	△̃	10000

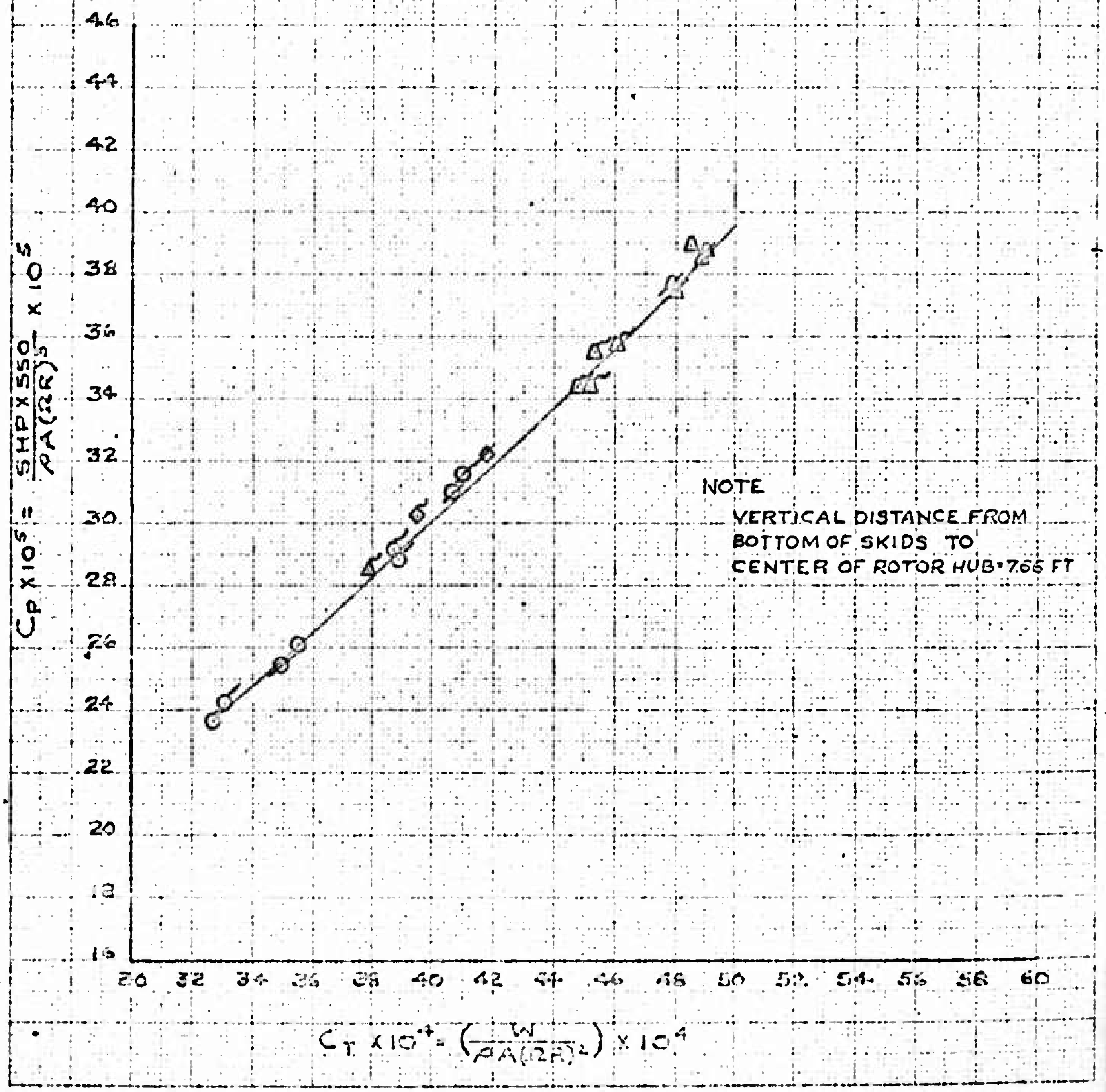
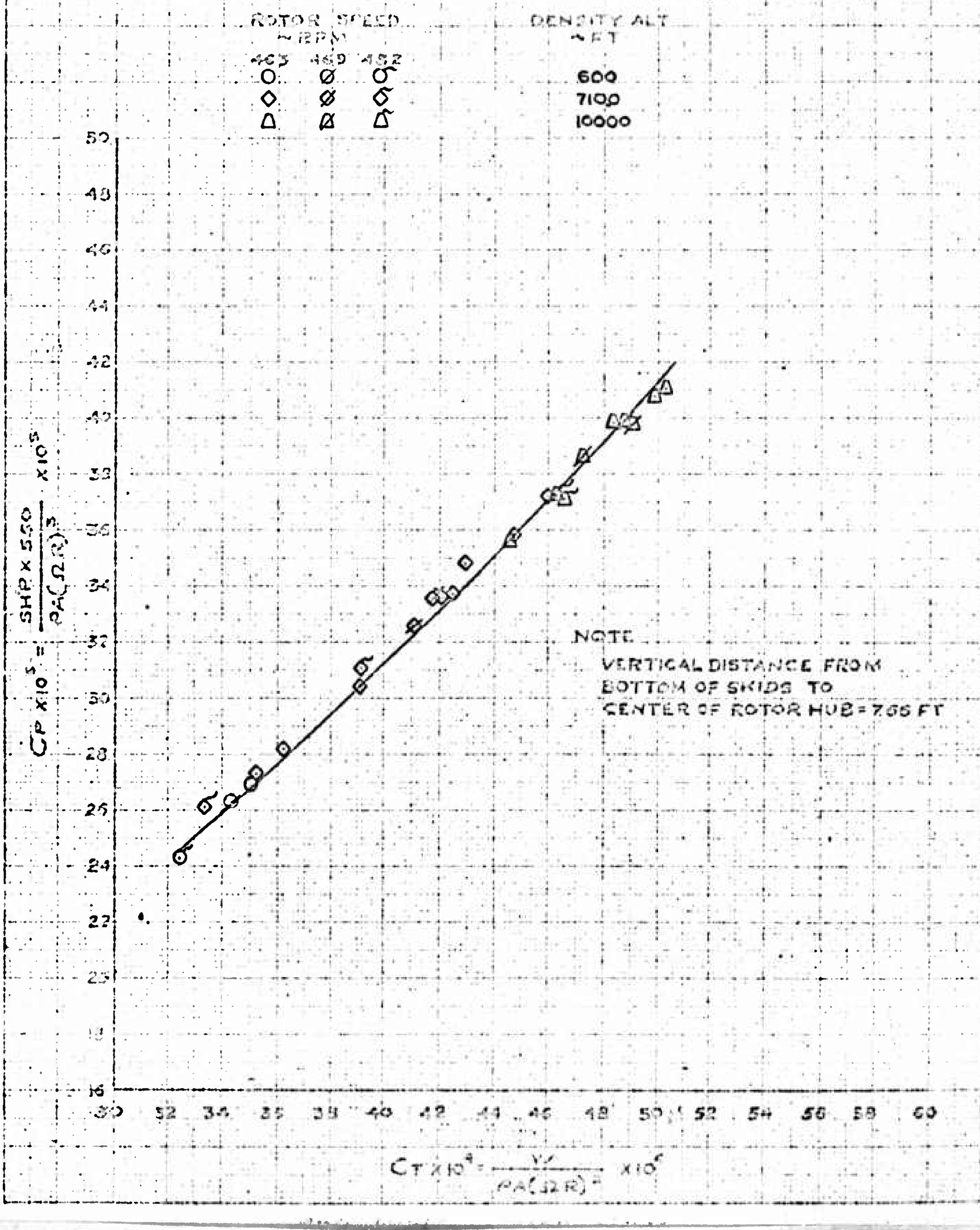


FIGURE No 18
HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT = 10 FT
CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS



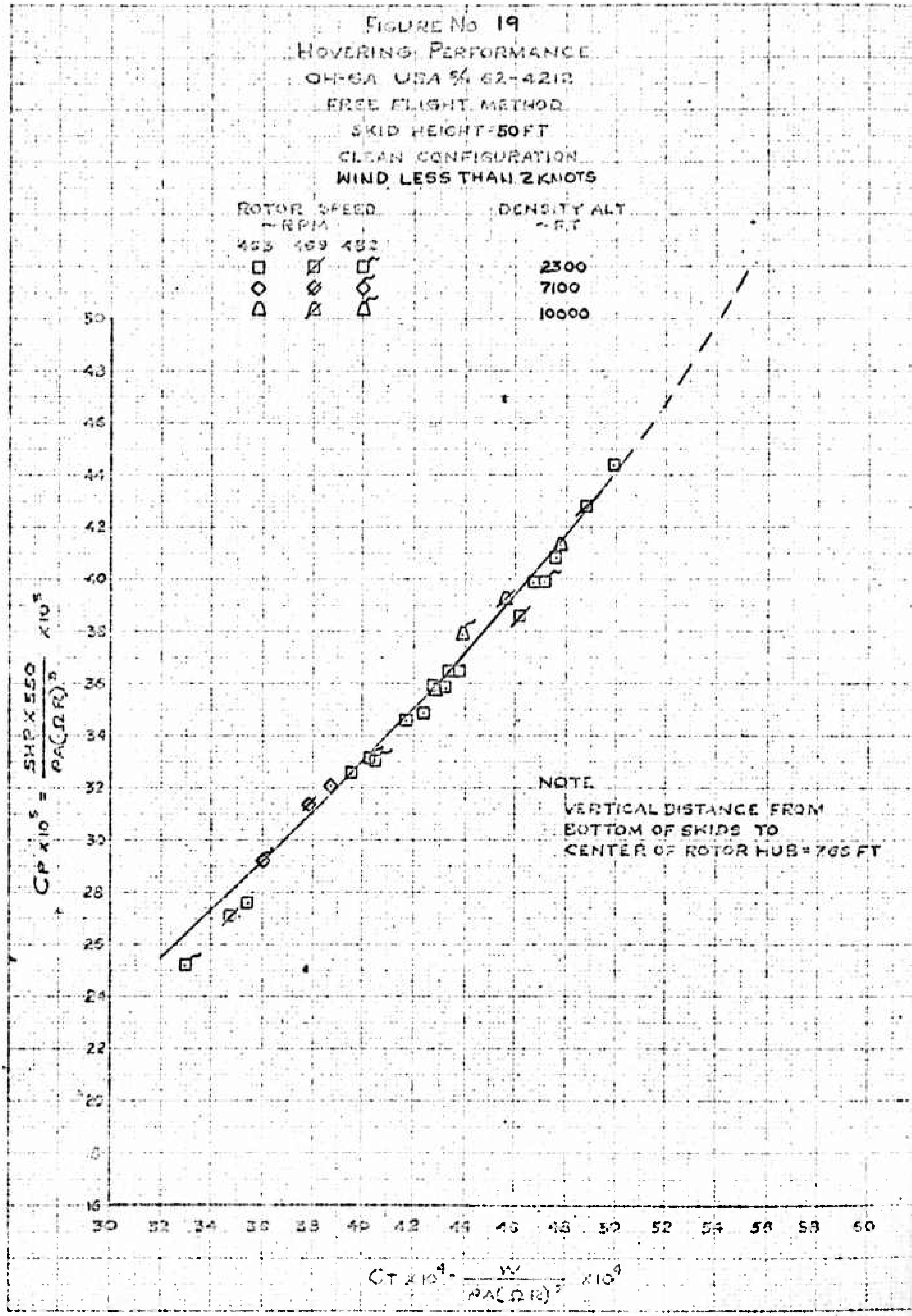
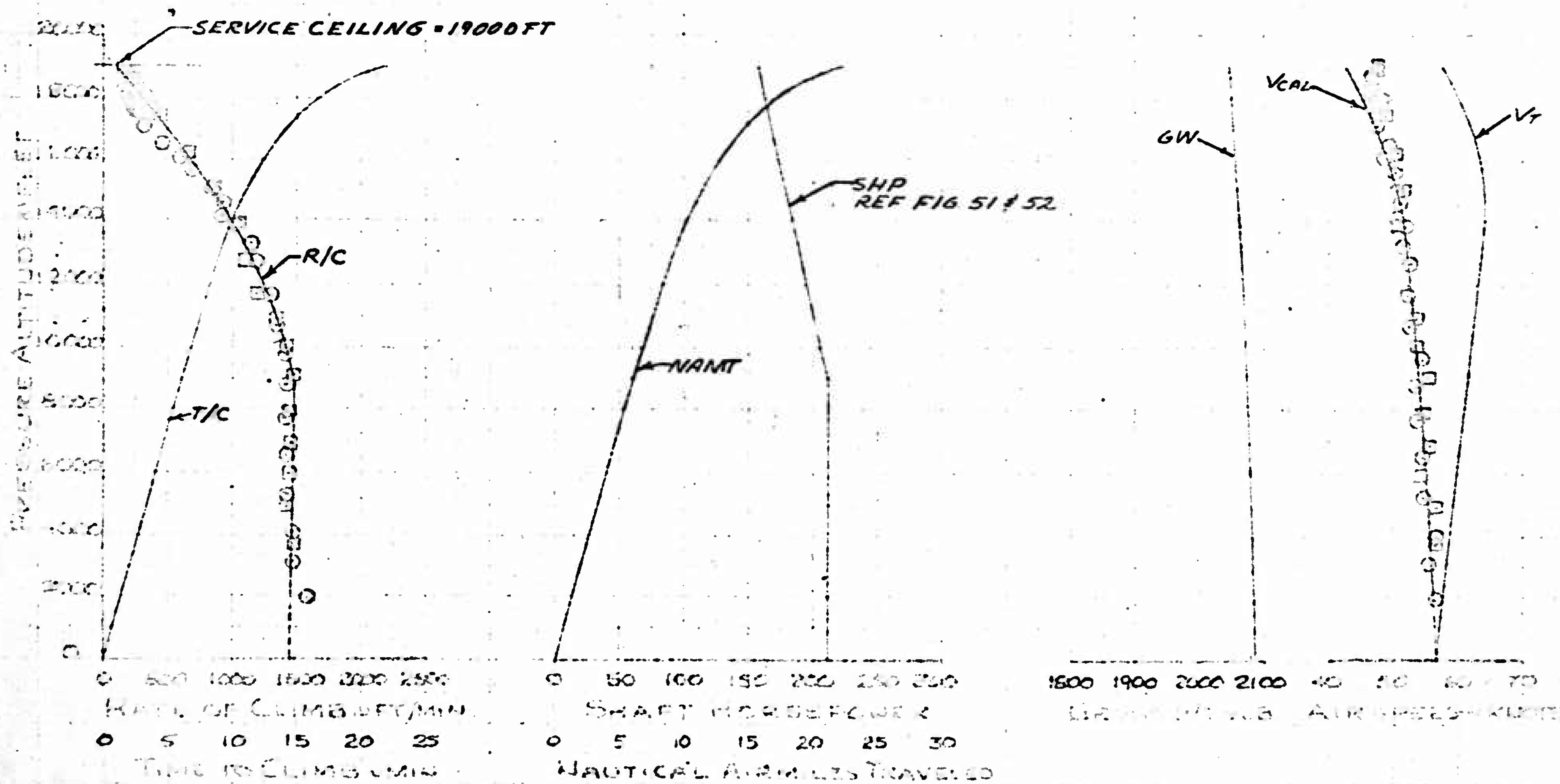


FIGURE No 19
HOVERING PERFORMANCE
OH-6A USA S/N 62-4212
FREE FLIGHT METHOD
SKID HEIGHT-50FT
CLEAN CONFIGURATION
WIND LESS THAN 2 KNOTS
ROTOR SPEED ~RPM
453 469 482
DENSITY ALT ~FT
2300
7100
10000
NOTE
VERTICAL DISTANCE FROM
BOTTOM OF SKIDS TO
CENTER OF ROTOR HUB = 7.65 FT
CP x10^5 = SHP X 550 / ρA(ΩR)^3 x10^5
CT x10^4 = W / ρA(ΩR)^2 x10^4
16
20
22
24
26
28
30
32
34
36
38
40
42
44
46
48
50
30
32
34
36
38
40
42
44
46
48
50
52
54
56
58
60

FIGURE NO. 20
CLIMB PERFORMANCE
OH-6A USA S/N 62-4212
MAXIMUM CONTINUOUS POWER
ROTOR SPEED 469 RPM
C.G. STATION NO. 100.3 (MID)
CLEAN CONFIGURATION
US. STANDARD DAY



1

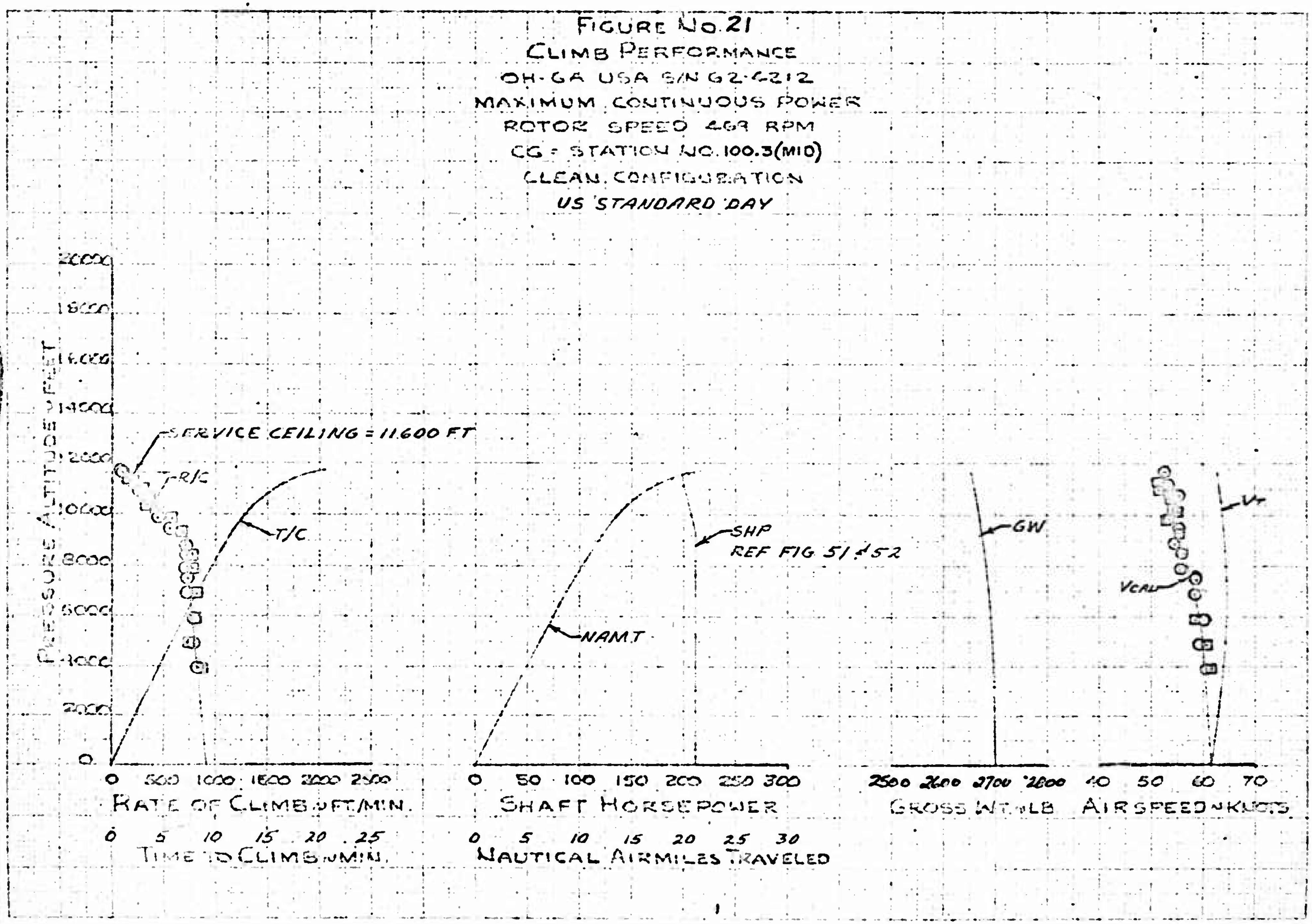
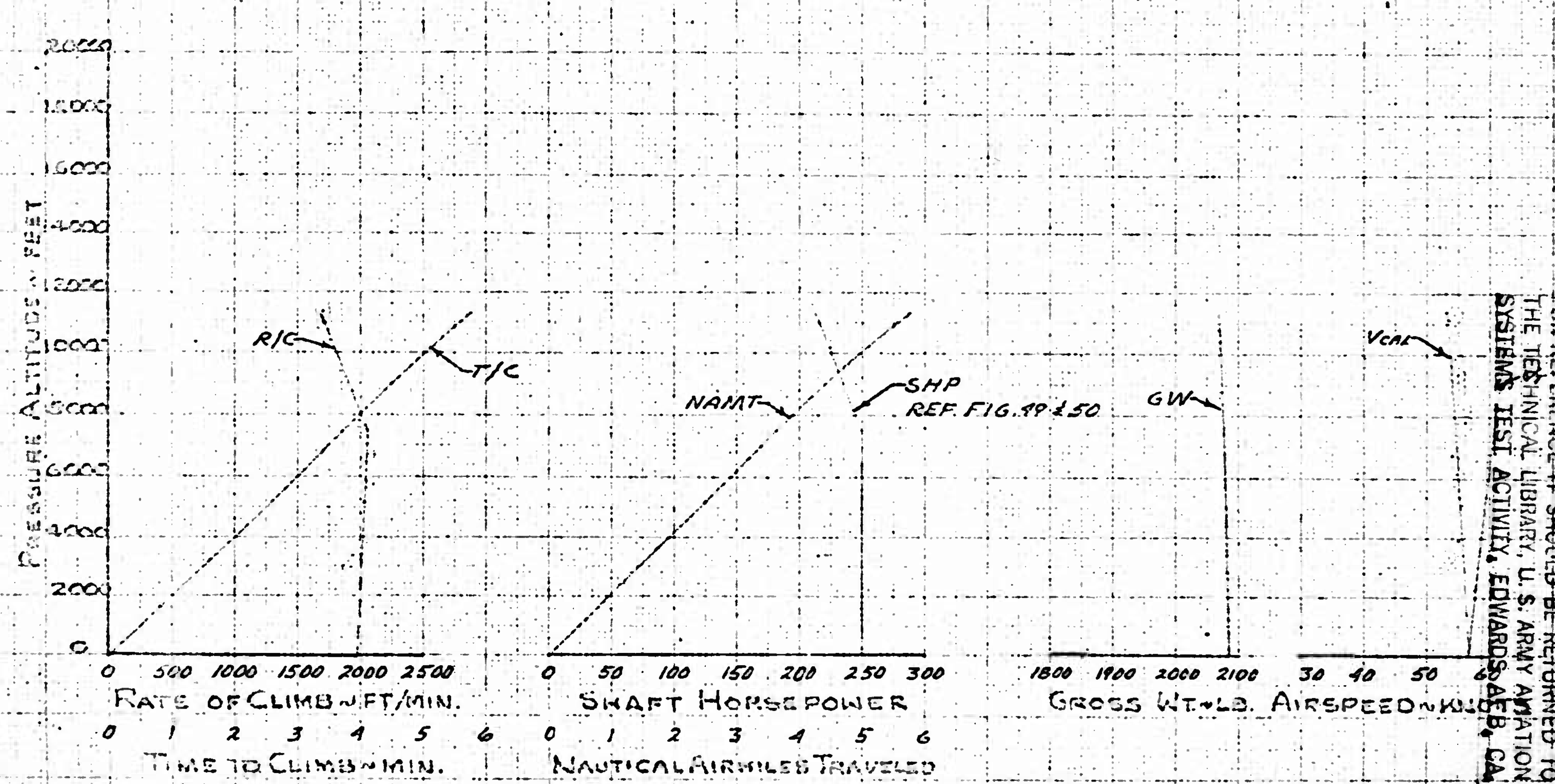


FIGURE NO. 21
CLIMB PERFORMANCE
OH-6A USA S/N 62-4212
MAXIMUM CONTINUOUS POWER
ROTOR SPEED 469 RPM
CG = STATION NO. 100.3 (MID)
CLEAN CONFIGURATION
US STANDARD DAY
PRESSURE ALTITUDE ~ FEET
0
2000
4000
6000
8000
10000
12000
14000
16000
18000
20000
SERVICE CEILING = 11,600 FT
R/C
T/C
NAMT
SHP
REF FIG 51 & 52
GW
VCAL
VT
0 500 1000 1500 2000 2500
RATE OF CLIMB ~ FT/MIN.
0 5 10 15 20 25
TIME TO CLIMB ~ MIN.
0 50 100 150 200 250 300
SHAFT HORSEPOWER
0 5 10 15 20 25 30
NAUTICAL AIRMILES TRAVELED
2500 2600 2700 2800
GROSS WT ~ LB
40 50 60 70
AIRSPEED ~ KNOTS

FIGURE NO. 22
CLIMB PERFORMANCE
OH-6A USA S/N 62-4212
TAKE-OFF POWER
ROTOR SPEED = 469 RPM
CG = STATION NO. 100.4 (MID)
CLEAN CONFIGURATION
US STANDARD DAY



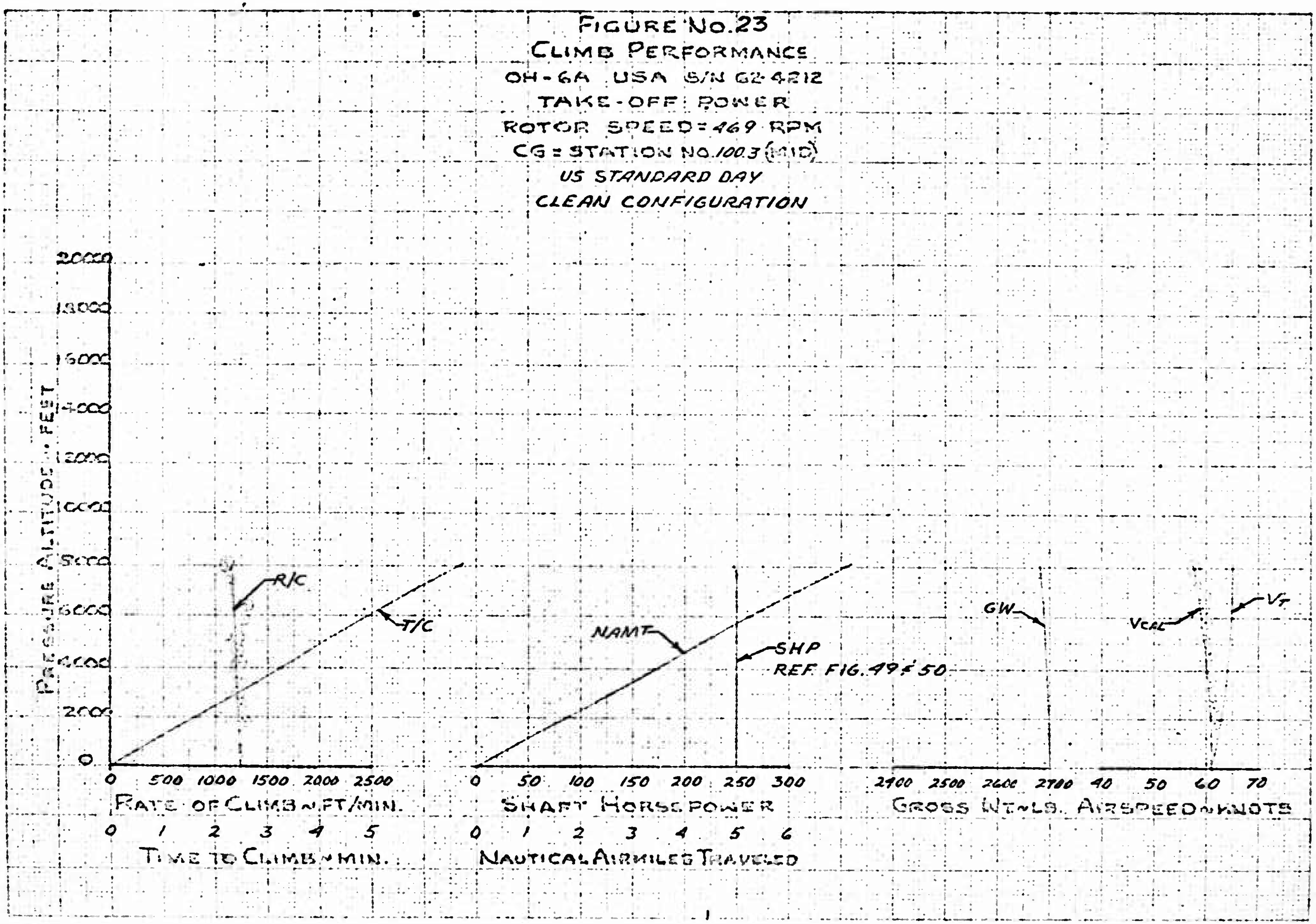
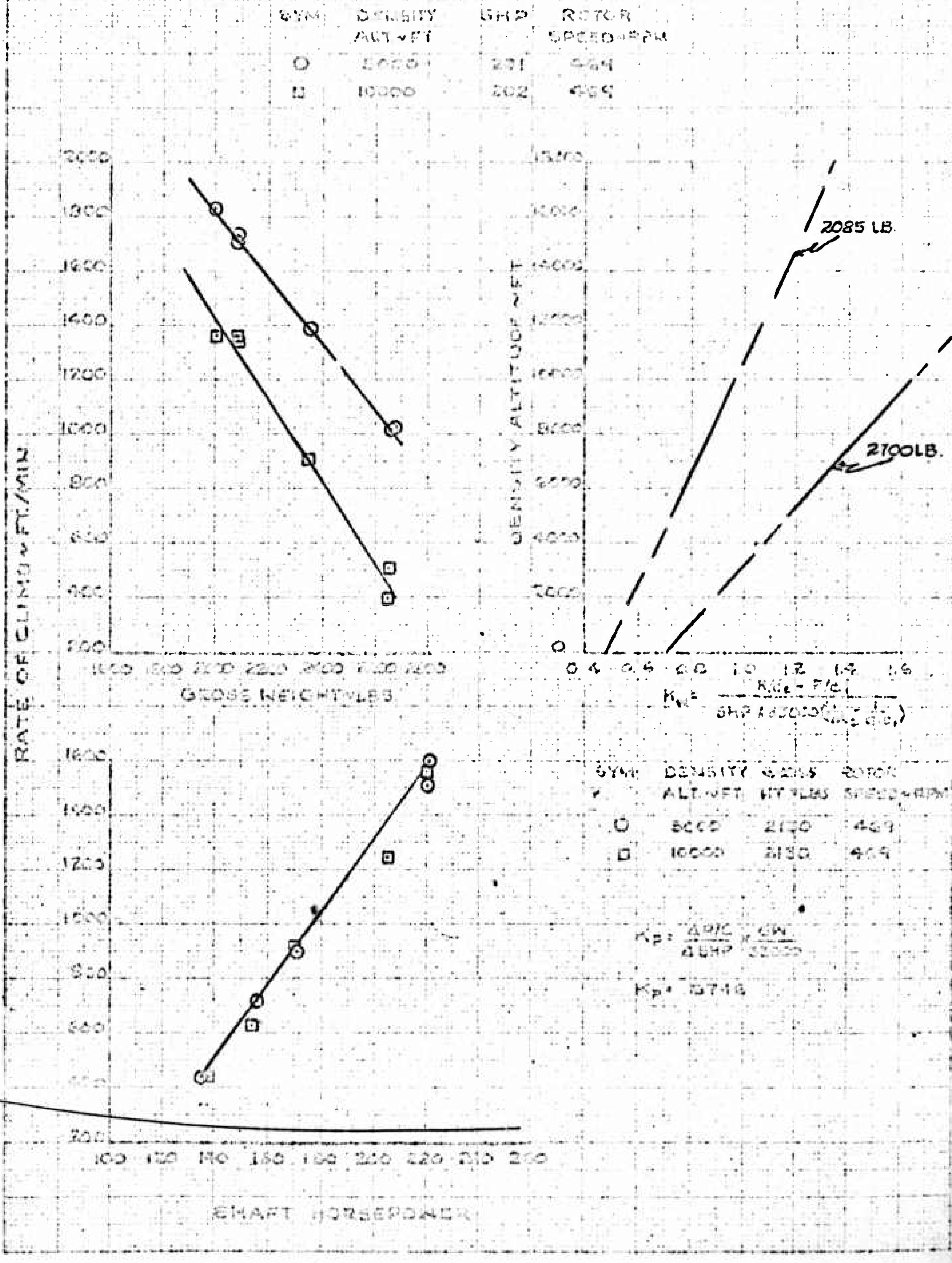


FIGURE NO. 23
CLIMB PERFORMANCE
OH-6A USA S/N 62-4212
TAKE-OFF POWER
ROTOR SPEED = 469 RPM
CG = STATION NO. 100.3 (MID)
US STANDARD DAY
CLEAN CONFIGURATION
PRESSURE ALTITUDE ~ FEET
20000
18000
16000
14000
12000
10000
8000
6000
4000
2000
0
R/C
T/C
NAMT
SHP
REF FIG. 49 & 50
GW
VCAL
VT
0 500 1000 1500 2000 2500
RATE OF CLIMB ~ FT/MIN.
0 1 2 3 4 5
TIME TO CLIMB ~ MIN.
0 50 100 150 200 250 300
SHAFT HORSEPOWER
0 1 2 3 4 5 6
NAUTICAL AIRMILES TRAVELED
2400 2500 2600 2700
GROSS WT ~ LB.
40 50 60 70
AIRSPEED ~ KNOTS

FIGURE No 24
VARIATION IN RATE OF CLIMB AS A FUNCTION OF GROSS WEIGHT AND SHAFT HORSEPOWER



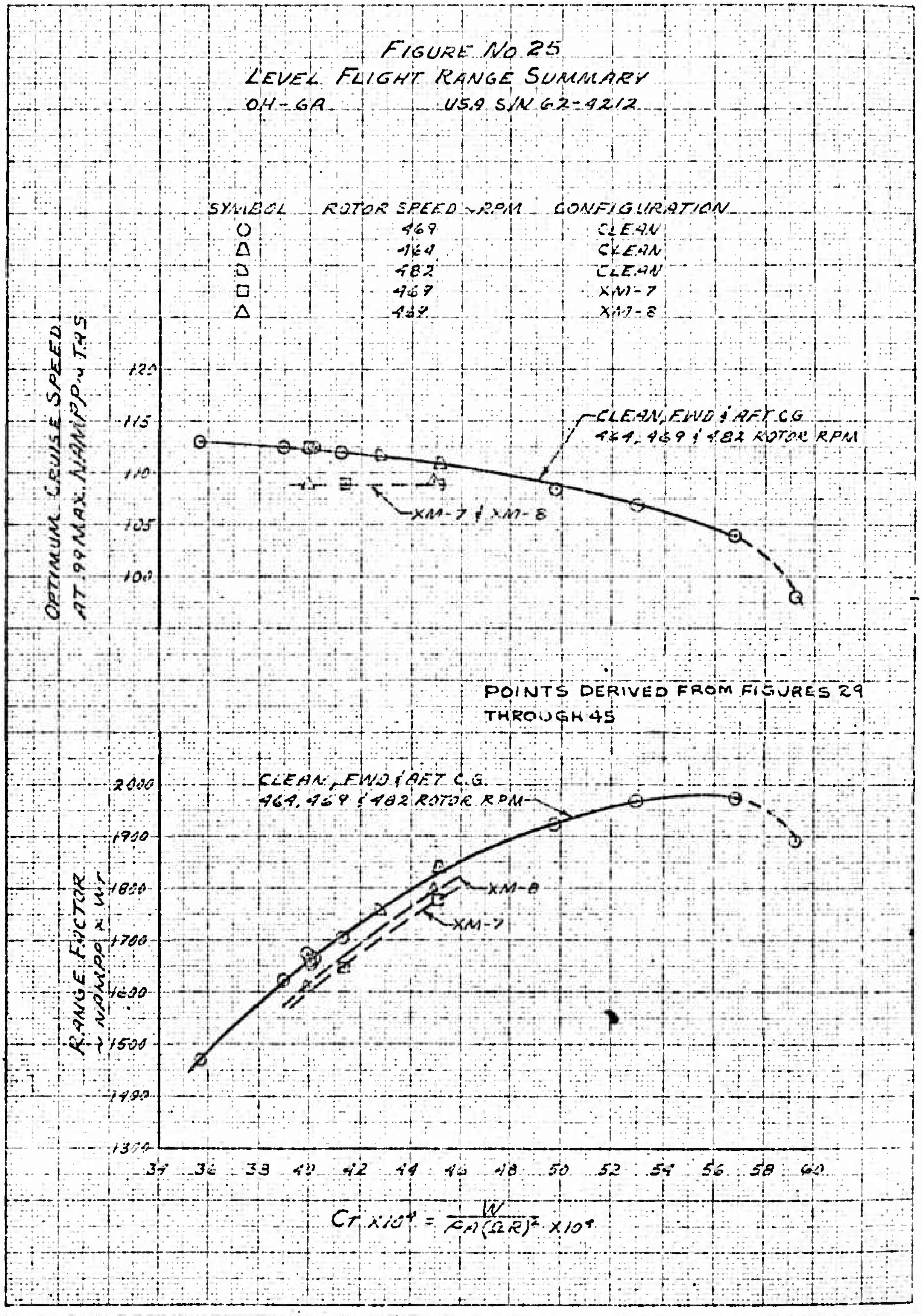


FIGURE NO 25
LEVEL FLIGHT RANGE SUMMARY
OH-6A USA S/N 62-4212

FIGURE No 26

LEVEL FLIGHT PERFORMANCE

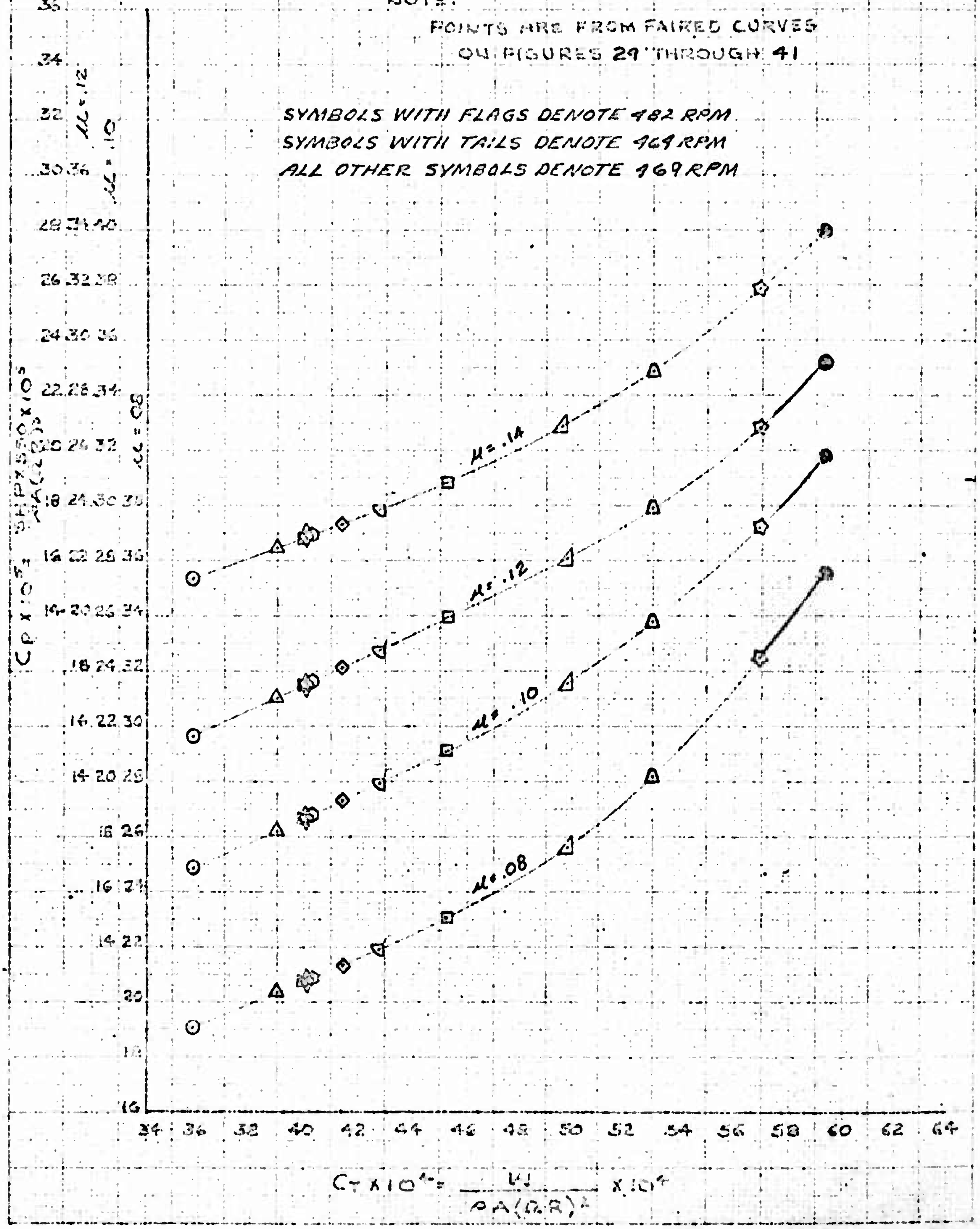
OH-6A USA S/N 62-4212

CG: STATION NO (MID)

CLEAN CONFIGURATION

NOTE:

POINTS ARE FROM FAIRED CURVES ON FIGURES 29 THROUGH 41



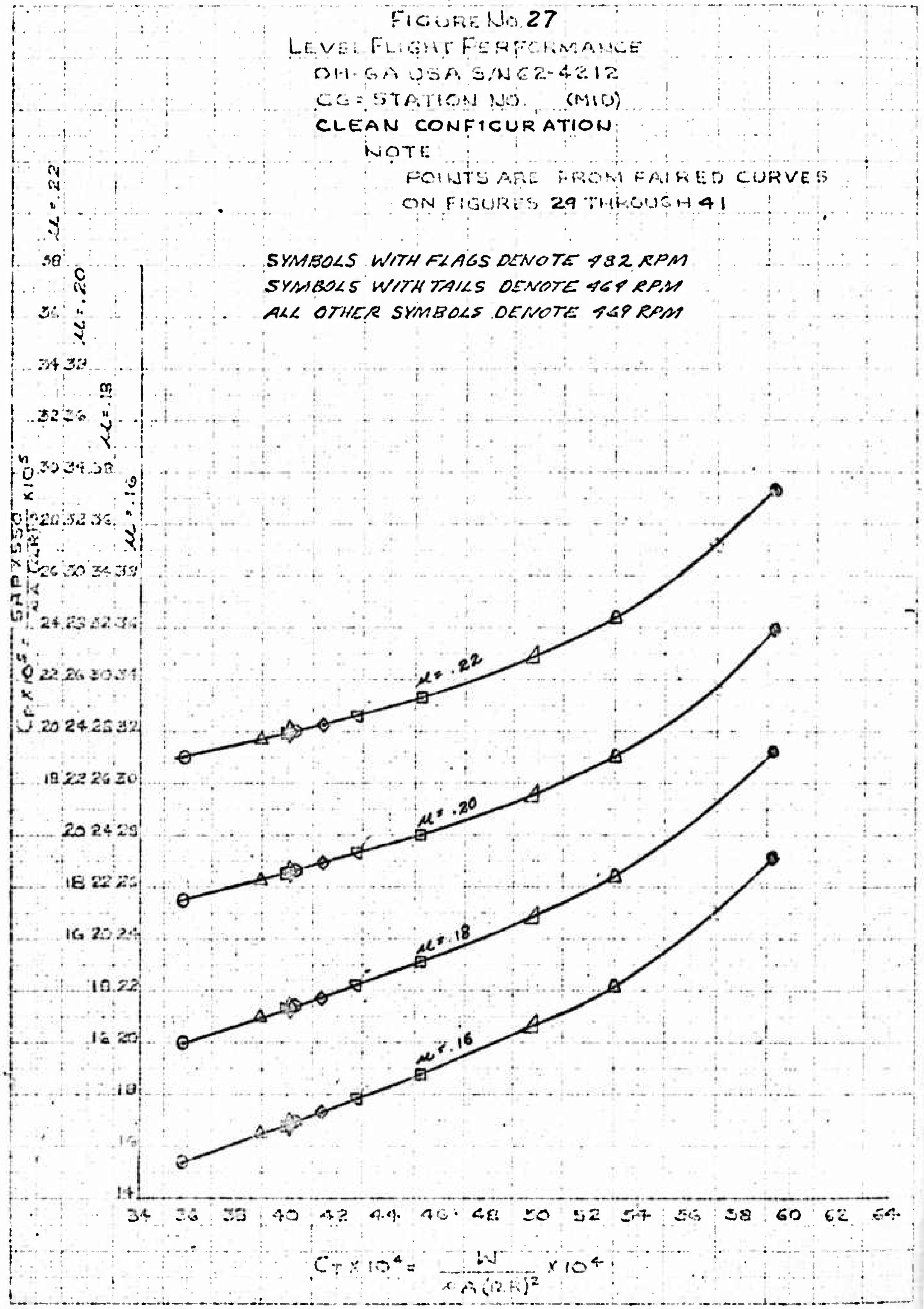


FIGURE No. 27
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
CG = STATION NO. (MID)
CLEAN CONFIGURATION
NOTE
POINTS ARE FROM FAIRED CURVES
ON FIGURES 29 THROUGH 41
SYMBOLS WITH FLAGS DENOTE 482 RPM
SYMBOLS WITH TAILS DENOTE 464 RPM
ALL OTHER SYMBOLS DENOTE 469 RPM
μ = .22
μ = .20
μ = .18
μ = .16
34 36 38 40 42 44 46 48 50 52 54 56 58 60 62 64
$C_T \times 10^4 = \frac{W}{\rho A (\Omega R)^2} \times 10^4$

FIGURE No. 28

LEVEL FLIGHT PERFORMANCE

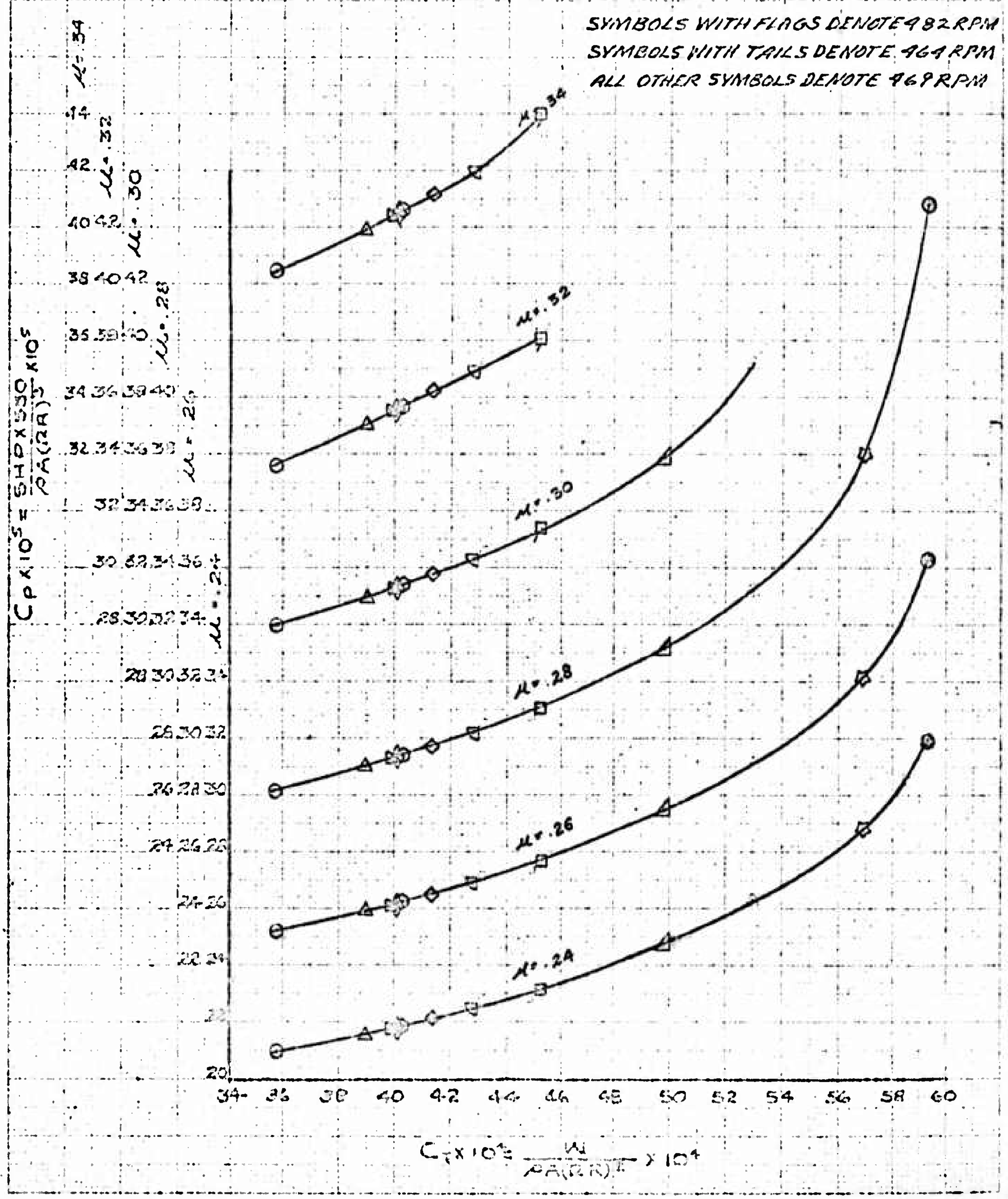
OH-6A USA S/N 62-4212

CG = STATION NO. (MID)

CLEAN CONFIGURATION

NOTE:

POINTS ARE FROM FAIRED CURVES ON FIGURES 29 THROUGH 41



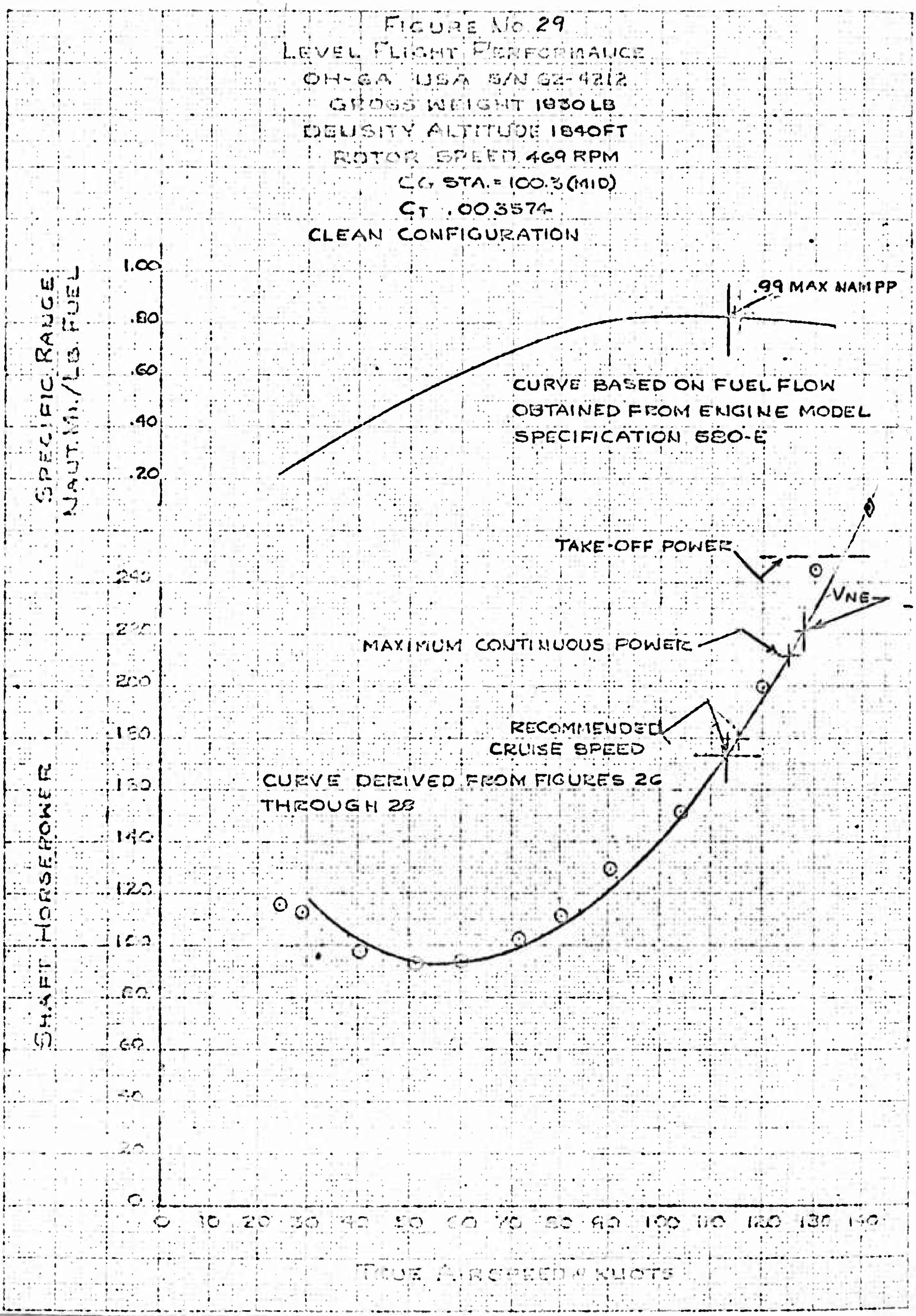


FIGURE No 29
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 1930 LB
DENSITY ALTITUDE 1840 FT
ROTOR SPEED 469 RPM
C.G. STA.= 100.3 (MID)
C_T .003574
CLEAN CONFIGURATION
SPECIFIC RANGE NAUT.MI./LB. FUEL
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 680-E
TAKE-OFF POWER
V_{NE}
MAXIMUM CONTINUOUS POWER
RECOMMENDED CRUISE SPEED
CURVE DERIVED FROM FIGURES 26 THROUGH 28
SHAFT HORSEPOWER
TRUE AIRSPEED ~ KNOTS

FIGURE No. 30
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2030 LB
DENSITY ALTITUDE 1260 FT
ROTOR SPEED 469 RPM
C.G. STA. = 100.3 (MID)
C_T .003887
CLEAN CONFIGURATION

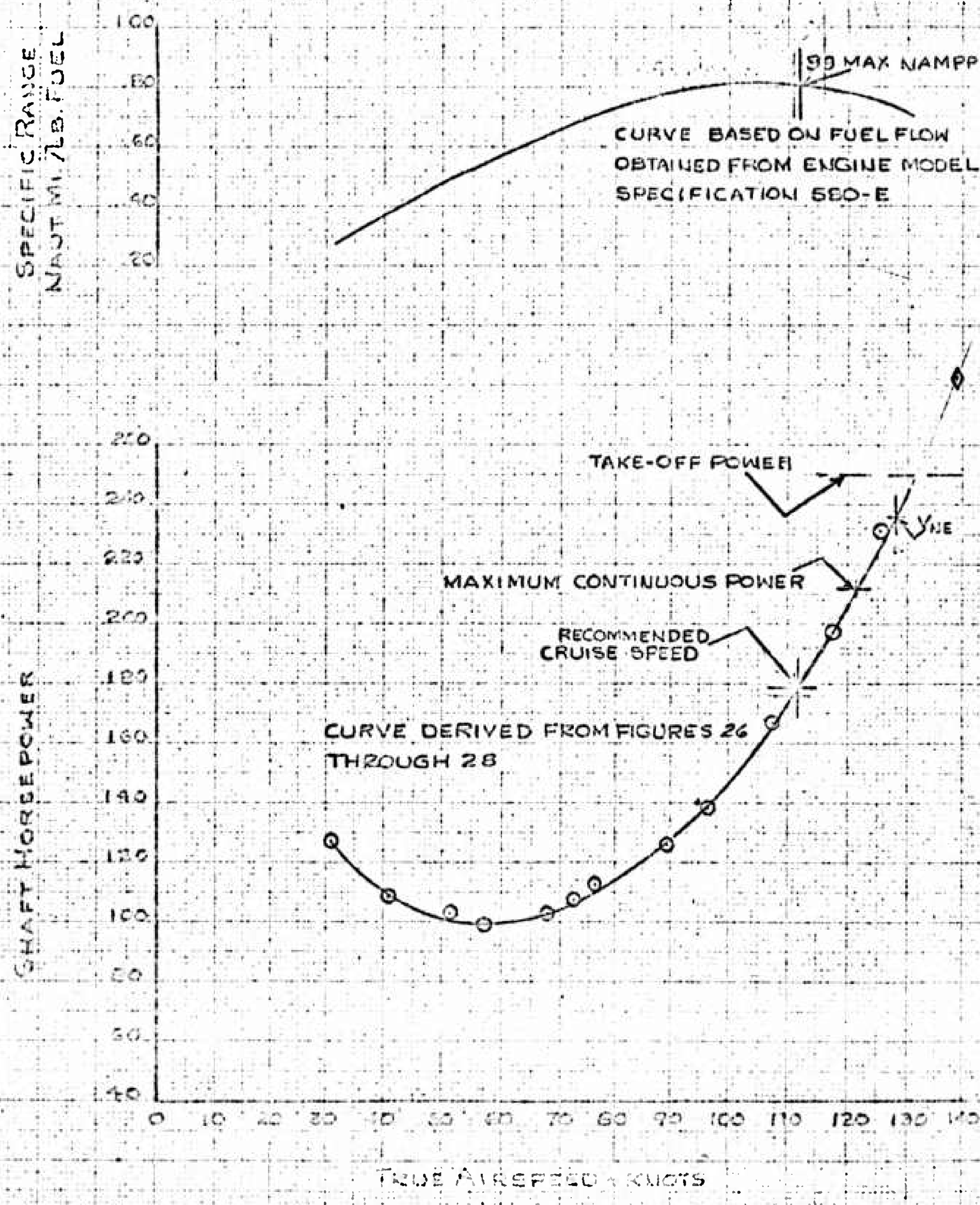
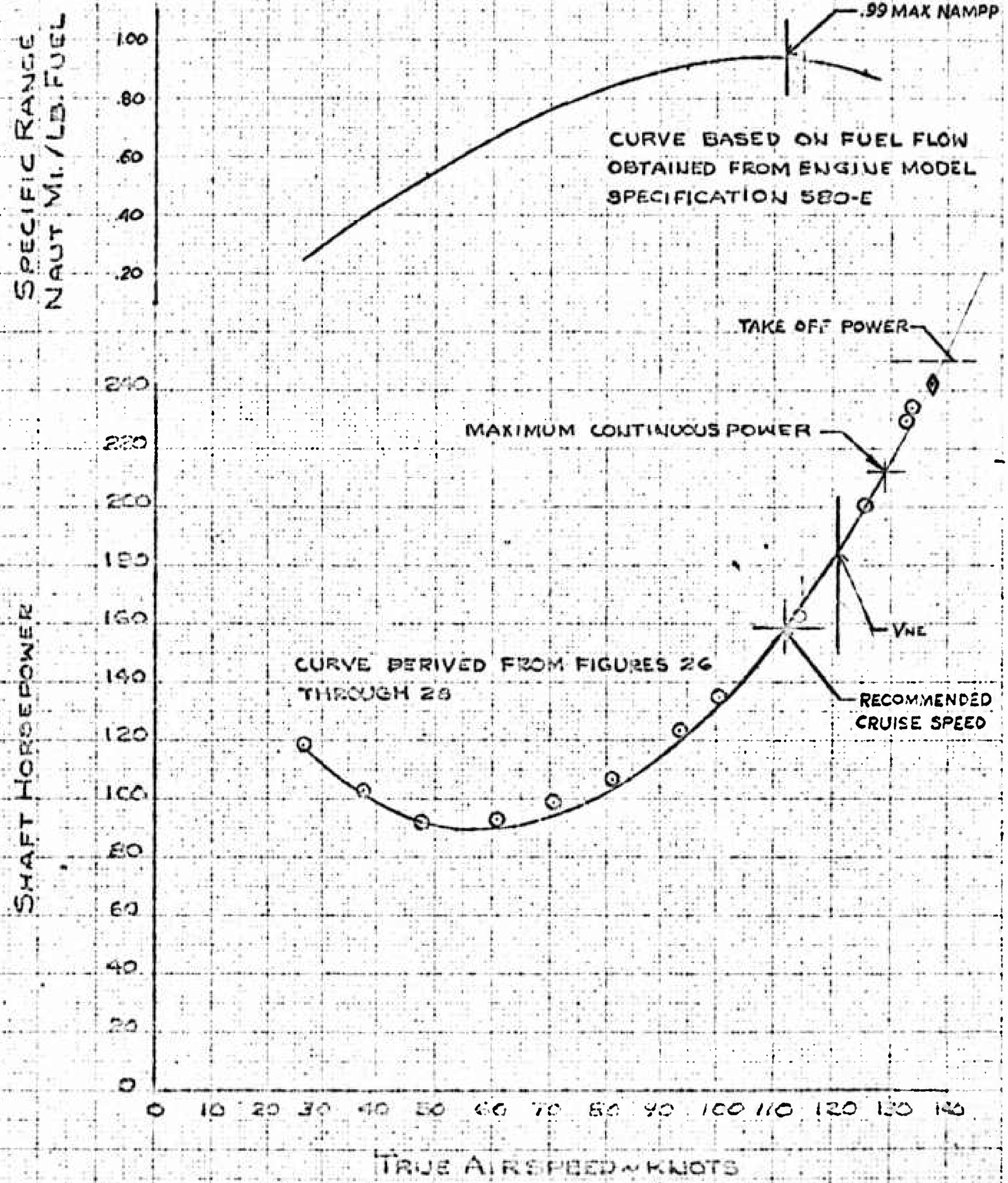


FIGURE NO 31
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 1820 LB
DENSITY ALTITUDE 5690 FT
ROTOR SPEED 469 RPM
C.G = STA 100.9 (MID)
C_T = .003950
CLEAN CONFIGURATION



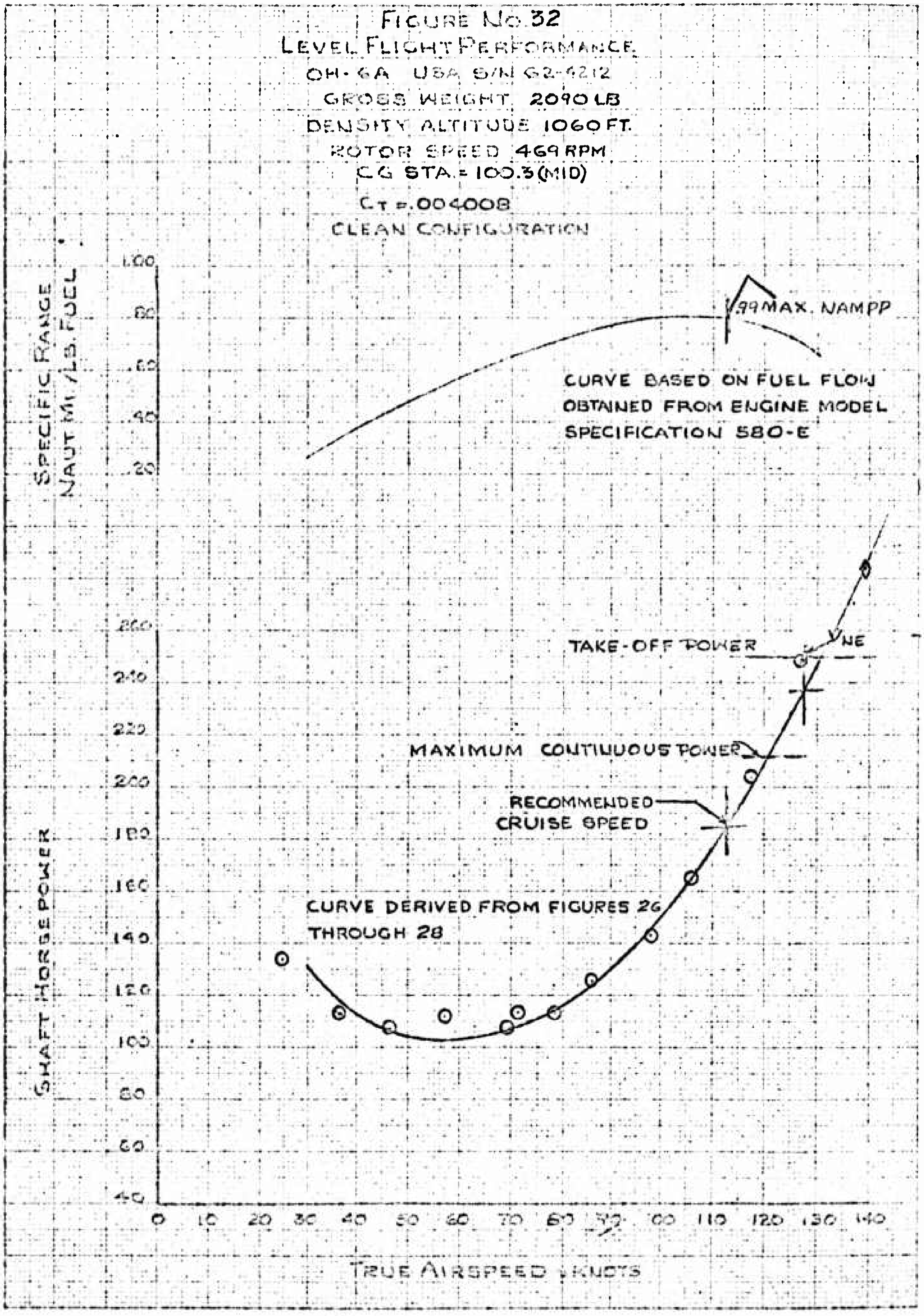
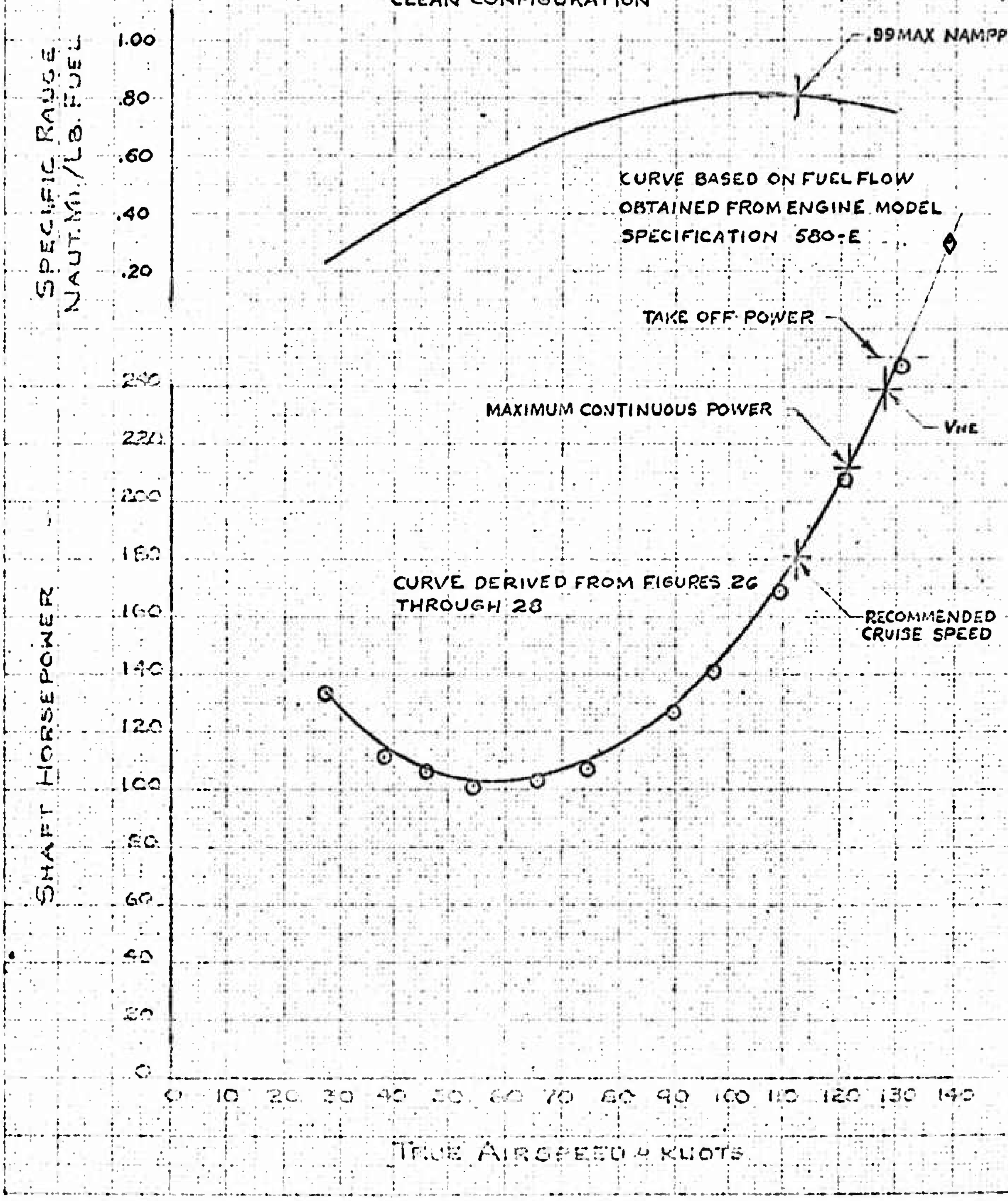


FIGURE No. 32
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2090 LB
DENSITY ALTITUDE 1060 FT.
ROTOR SPEED 469 RPM
C.G STA. = 100.3 (MID)
CT = .004008
CLEAN CONFIGURATION
SPECIFIC RANGE
NAUT MI./LB. FUEL
1.00
.80
.60
.40
.20
.99 MAX. NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
SHAFT HORSEPOWER
260
240
220
200
180
160
140
120
100
80
60
40
TAKE-OFF POWER
VNE
MAXIMUM CONTINUOUS POWER
RECOMMENDED CRUISE SPEED
CURVE DERIVED FROM FIGURES 26 THROUGH 28
0 10 20 30 40 50 60 70 80 90 100 110 120 130 140
TRUE AIRSPEED KNOTS

FIGURE No. 33
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2080 LB
DENSITY ALTITUDE 1490 FT.
ROTOR SPEED 469 RPM
C.G.= STA 100.5 (MID)
C_T = .004030
CLEAN CONFIGURATION



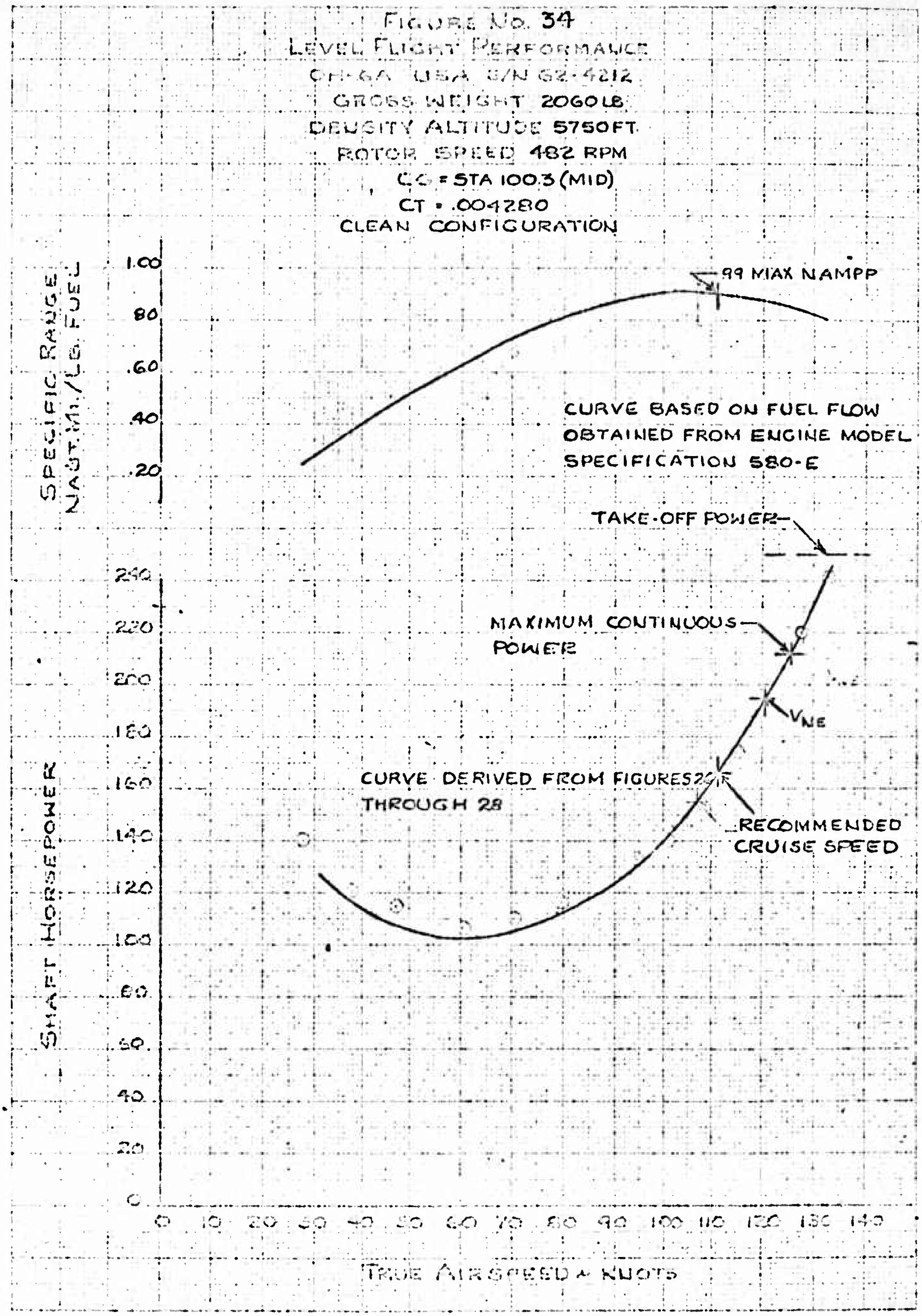


FIGURE No. 34
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2060 LB
DENSITY ALTITUDE 5750 FT
ROTOR SPEED 482 RPM
C.G. = STA 100.3 (MID)
CT = .004280
CLEAN CONFIGURATION
SPECIFIC RANGE
NAUT. MI./LB. FUEL
1.00
.80
.60
.40
.20
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
TAKE-OFF POWER
MAXIMUM CONTINUOUS POWER
VNE
CURVE DERIVED FROM FIGURES 24 THROUGH 28
RECOMMENDED CRUISE SPEED
SHAFT HORSEPOWER
240
220
200
180
160
140
120
100
80
60
40
20
0
0 10 20 30 40 50 60 70 80 90 100 110 120 130 140
TRUE AIRSPEED ~ KNOTS

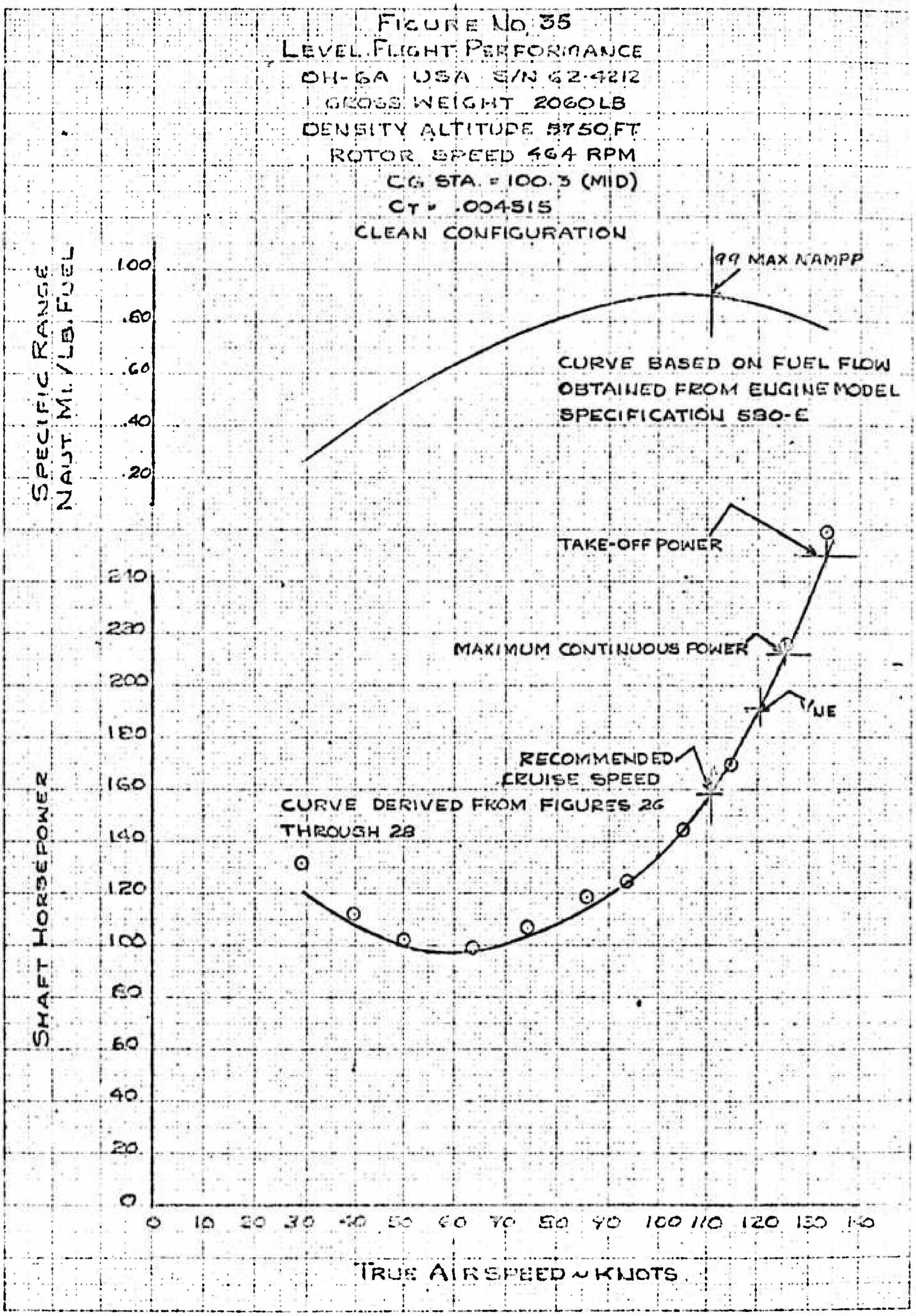
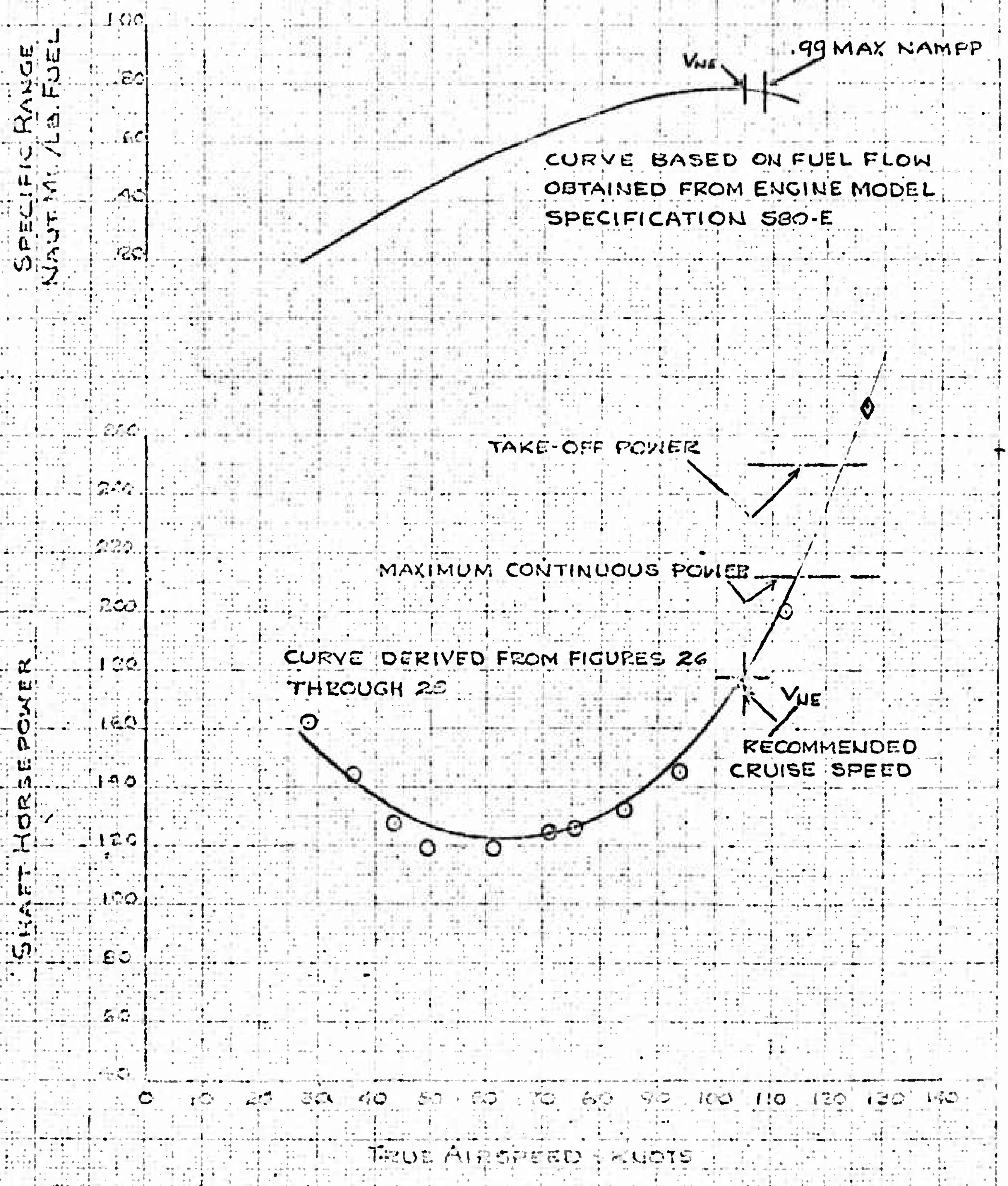


FIGURE NO. 35
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2060 LB
DENSITY ALTITUDE 5750 FT
ROTOR SPEED 464 RPM
C.G. STA. = 100.3 (MID)
CT = .004515
CLEAN CONFIGURATION
SPECIFIC RANGE
NAUT. MI./LB. FUEL
1.00
.80
.60
.40
.20
99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
TAKE-OFF POWER
MAXIMUM CONTINUOUS POWER
VNE
RECOMMENDED CRUISE SPEED
CURVE DERIVED FROM FIGURES 26 THROUGH 28
SHAFT HORSEPOWER
240
220
200
180
160
140
120
100
80
60
40
20
0
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
TRUE AIRSPEED ~ KNOTS

FIGURE No. 36
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2500 LB.
DENSITY ALTITUDE 2510 FT.
ROTOR SPEED 469 RPM
C.G. STA. = 100.3 (MID)
C_T = .004987
CLEAN CONFIGURATION



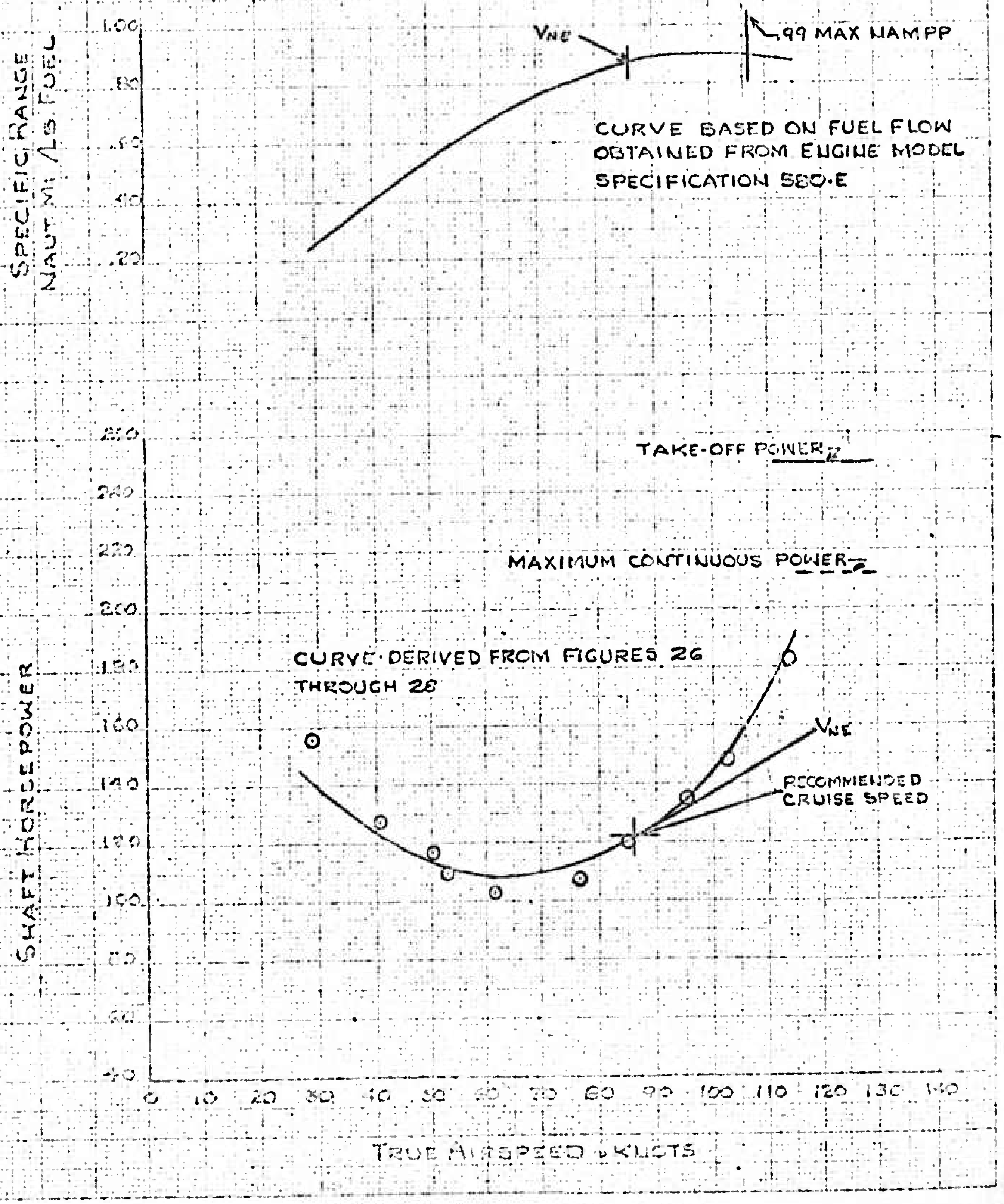


FIGURE No. 37

LEVEL FLIGHT PERFORMANCE

OH-6A USA S/N 62-4212

GROSS WEIGHT 2200 LB

DENSITY ALTITUDE 8740 FT

ROTOR SPEED 469 RPM

CG STA. = 100.4 (MID)

C_T .005300

CLEAN CONFIGURATION

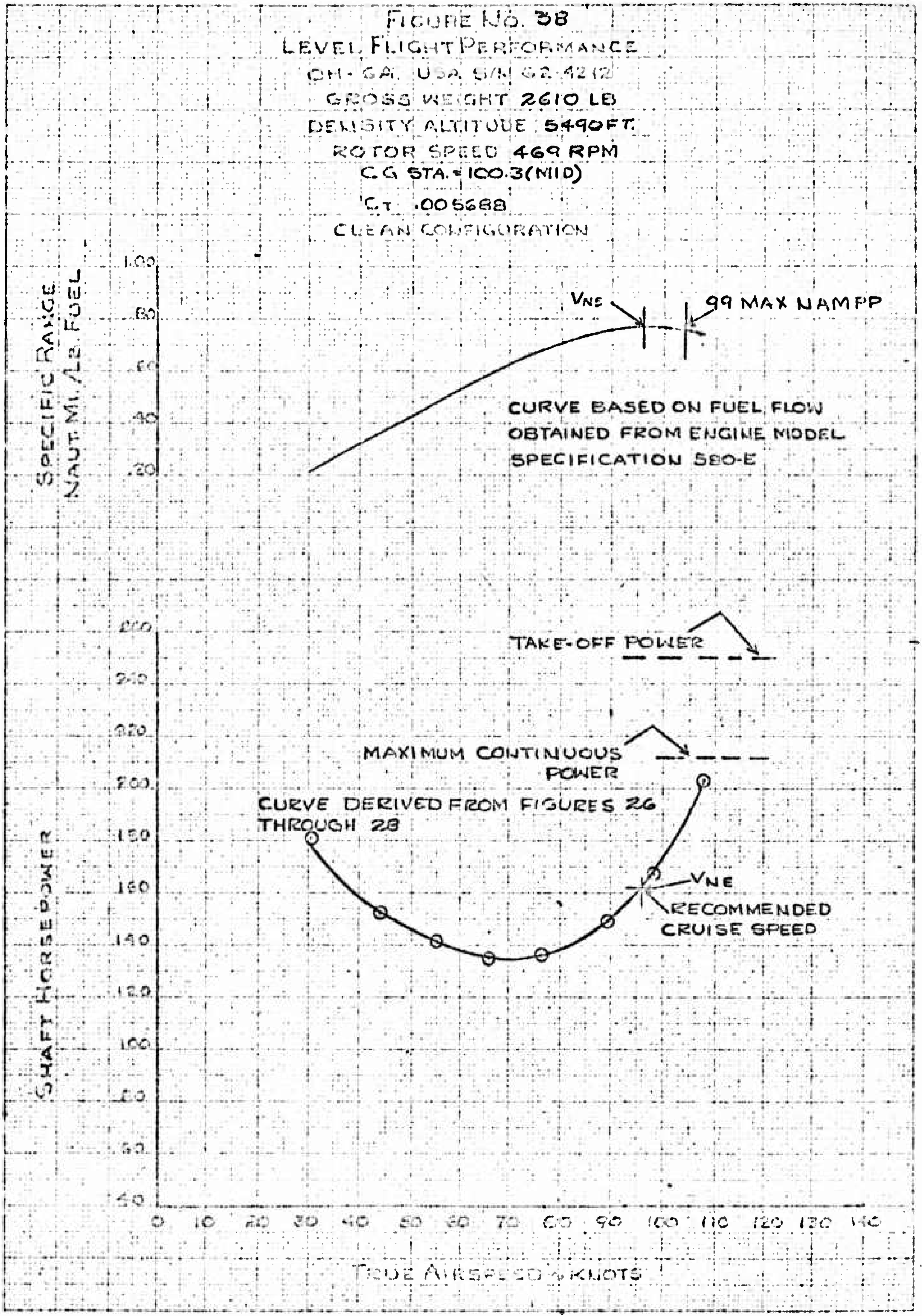


FIGURE NO. 38
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2610 LB
DENSITY ALTITUDE 5490 FT.
ROTOR SPEED 469 RPM
C.G STA. = 100.3 (MID)
C_T .005688
CLEAN CONFIGURATION
SPECIFIC RANGE
NAUT MI / LB FUEL
1.00
.80
.60
.40
.20
VNE
99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
TAKE-OFF POWER
MAXIMUM CONTINUOUS POWER
CURVE DERIVED FROM FIGURES 26 THROUGH 28
VNE
RECOMMENDED CRUISE SPEED
SHAFT HORSEPOWER
260
240
220
200
180
160
140
120
100
80
60
40
0 10 20 30 40 50 60 70 80 90 100 110 120 130 140
TRUE AIRSPEED ~ KNOTS

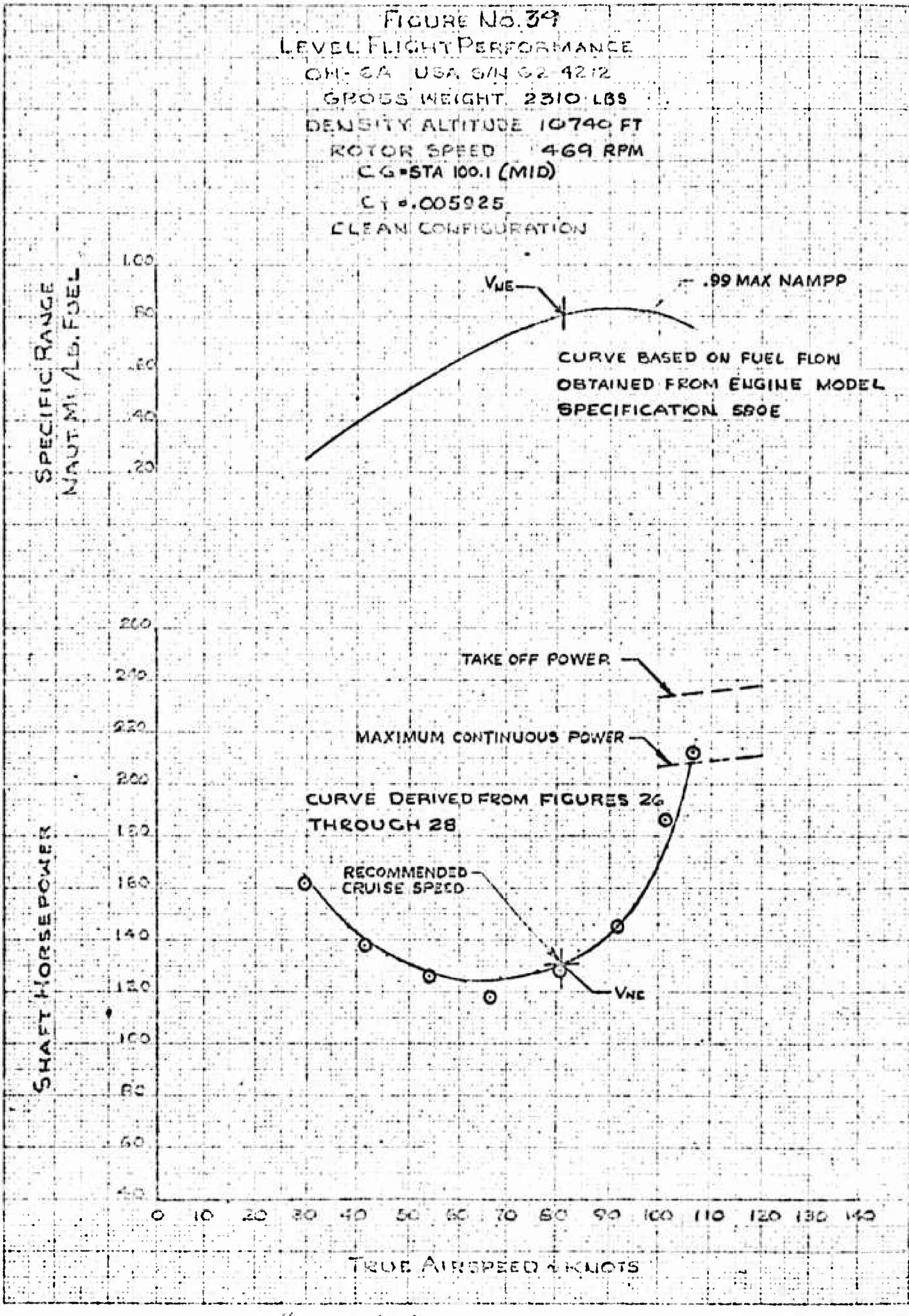
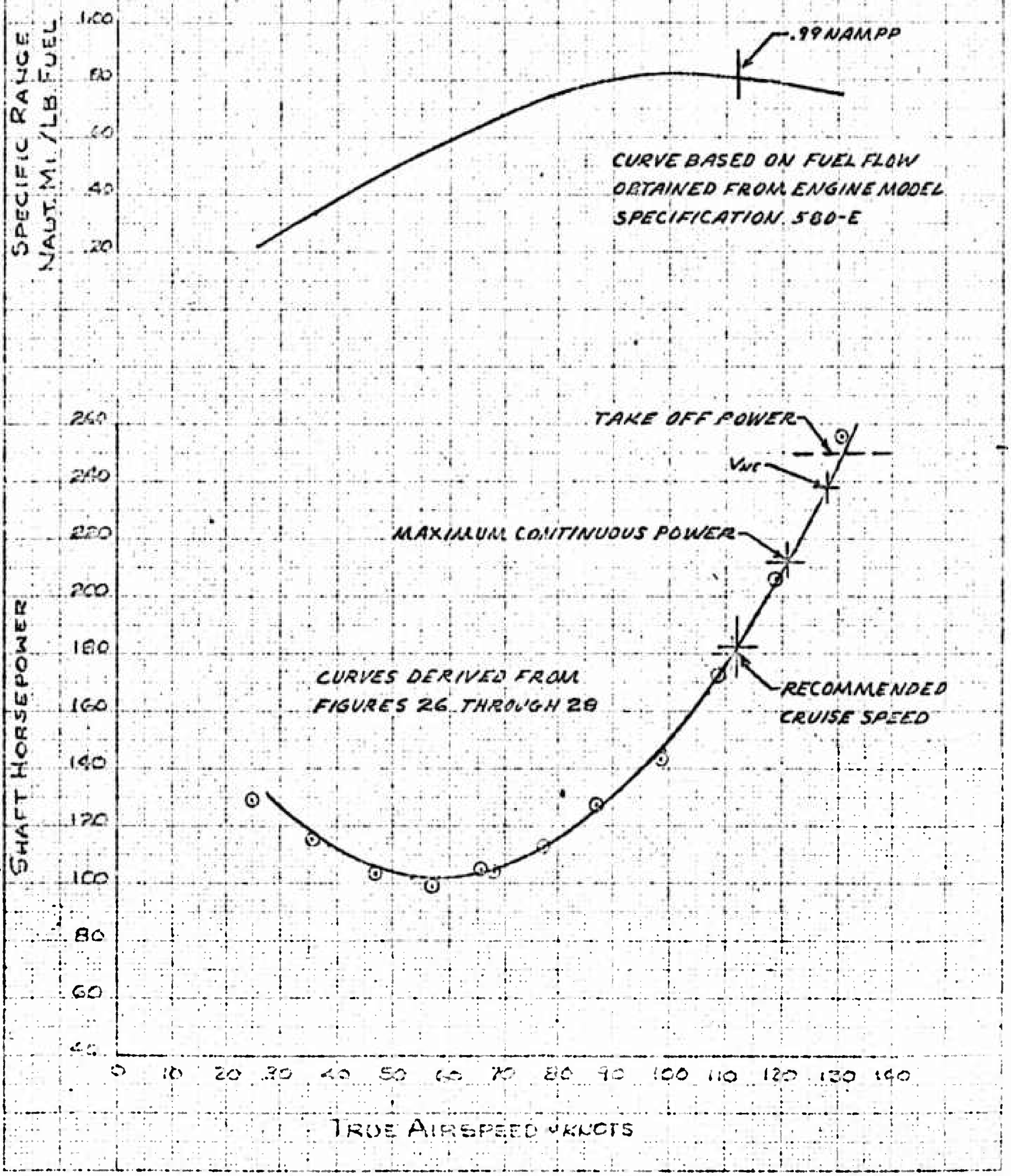
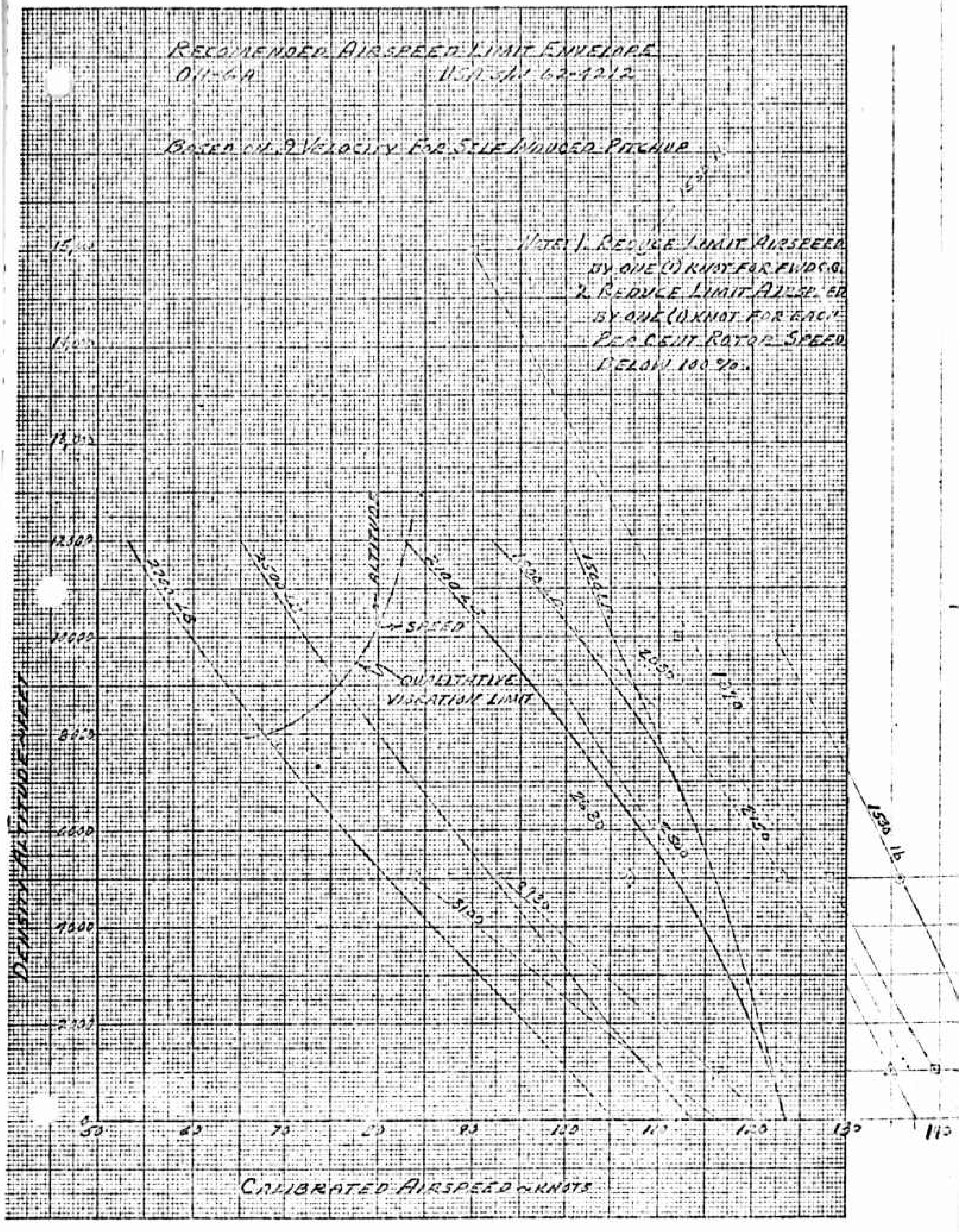


FIGURE No. 39
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2310 LBS
DENSITY ALTITUDE 10740 FT
ROTOR SPEED 469 RPM
C.G = STA 100.1 (MID)
CT = .005925
CLEAN CONFIGURATION
SPECIFIC RANGE
NAUT MI / LB. FUEL
VNE
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580E
SHAFT HORSEPOWER
TAKE OFF POWER
MAXIMUM CONTINUOUS POWER
CURVE DERIVED FROM FIGURES 26 THROUGH 28
RECOMMENDED CRUISE SPEED
VNE
TRUE AIRSPEED ~ KNOTS

FIGURE No 40
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2070 LB
DENSITY ALTITUDE 1600 FT
ROTOR SPEED 469 RPM
CG = 97.0 (FWD)
C_T = .004012
CLEAN CONFIGURATION





RECOMENDED AIRSPEED LIMIT ENVELOPE
OH-6A
USA S/N 62-4212
BASED ON 9 VELOCITY FOR SELF INDUCED PITCHUP
NOTE: 1. REDUCE LIMIT AIRSPEED BY ONE (1) KNOT FOR FWD C.G.
2. REDUCE LIMIT AIRSPEED BY ONE (1) KNOT FOR EACH PERCENT ROTOR SPEED BELOW 100 %.
ALTITUDE
SPEED
QUALITATIVE VIBRATION LIMIT
2500 LB
2000 LB
1500 LB
1530 lb
DENSITY ALTITUDE ~ FEET
CALIBRATED AIRSPEED ~ KNOTS
50
60
70
80
90
100
110
120
130
140

FIGURE No 41
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2090 LB
DENSITY ALTITUDE 2310 FT
ROTOR SPEED 469 RPM
CG = STA 103 (AFT)
C_T .004139
CLEAN CONFIGURATION

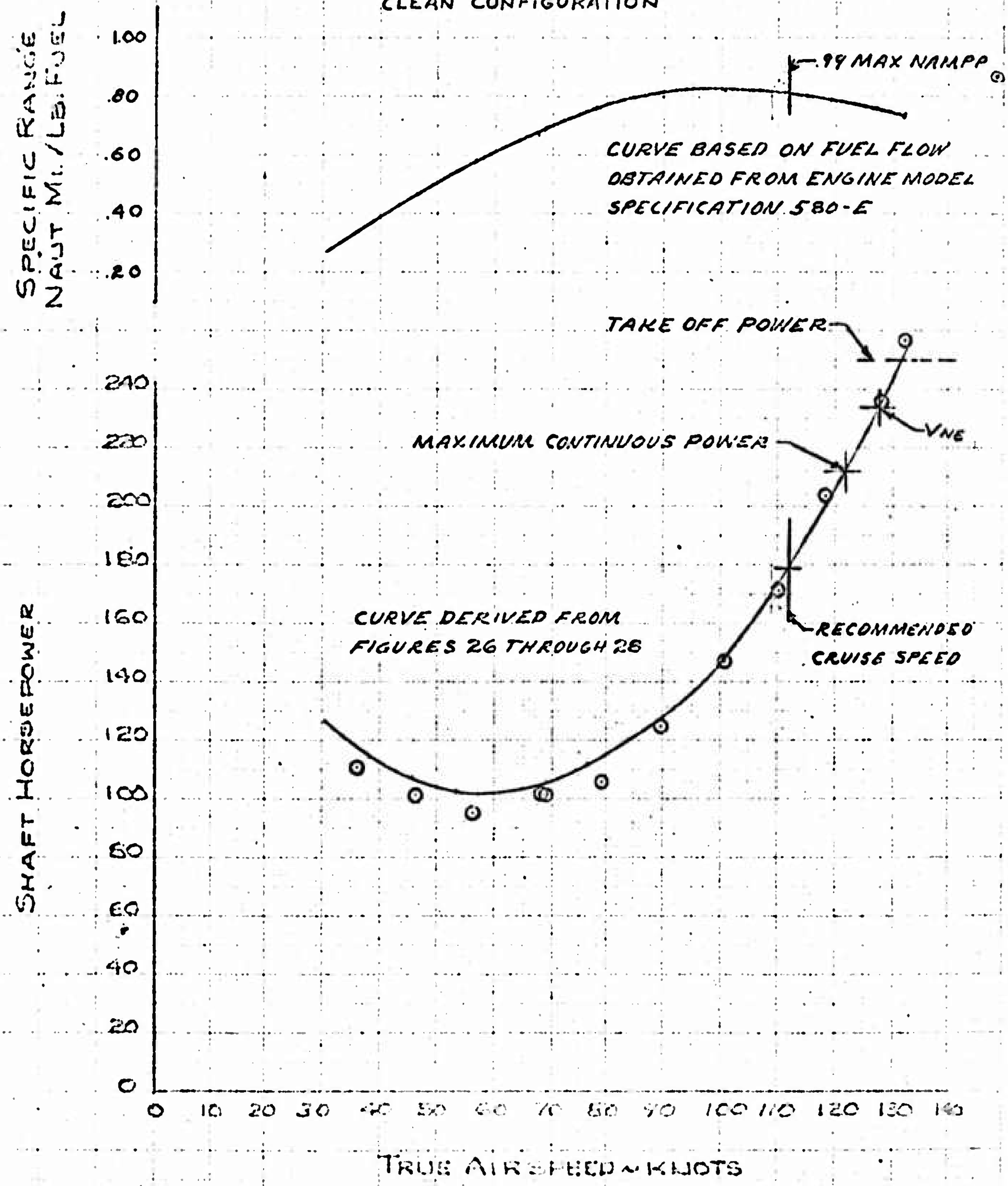
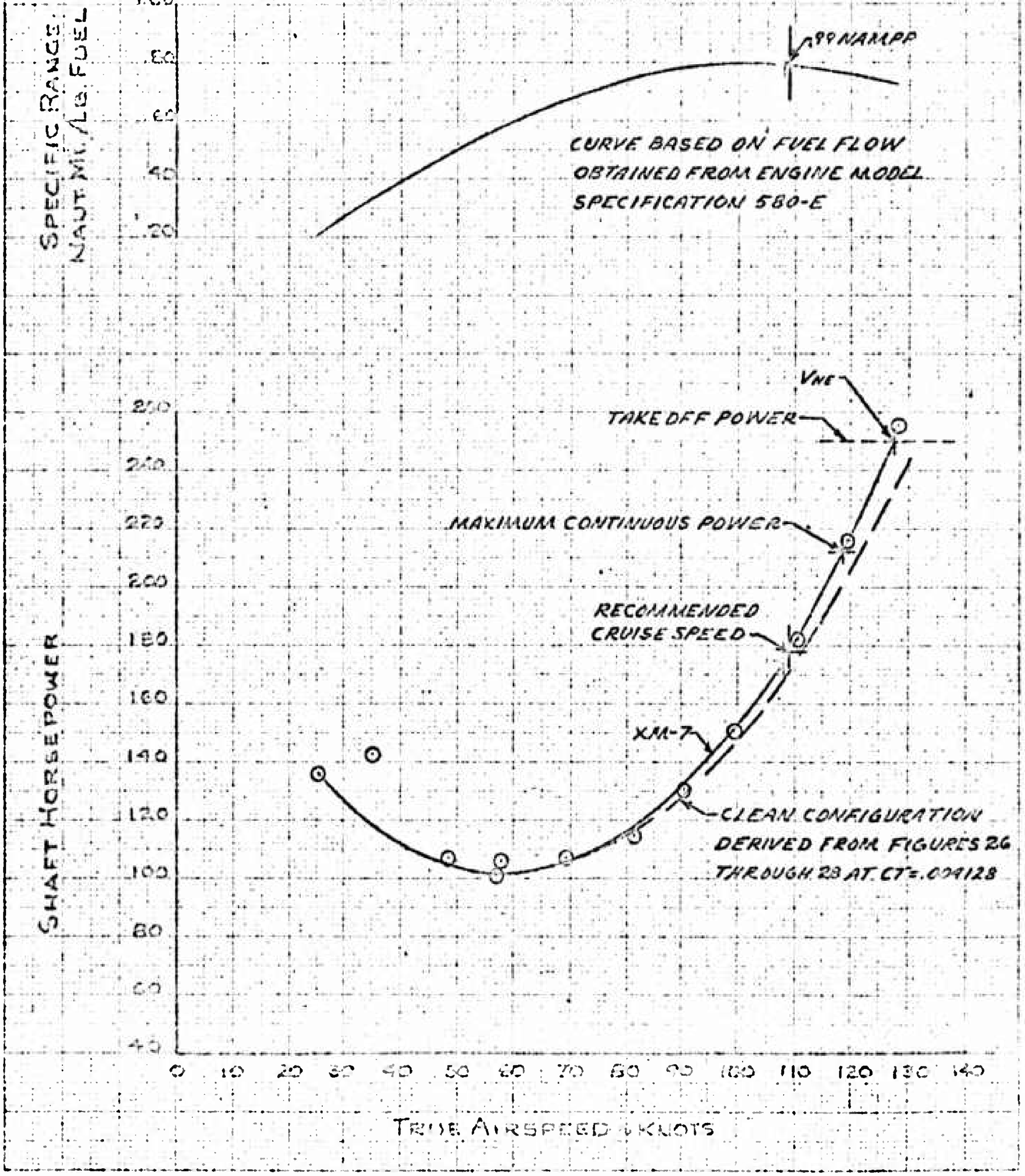


FIGURE NO. 42
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2085 LB.
DENSITY ALTITUDE 2300 FT
ROTOR SPEED 469 RPM
C G = STA 100.2 (MID)
C_T = .004128
XM-7 ARMAMENT SYSTEM
INSTALLED IN STANDARD POSITION



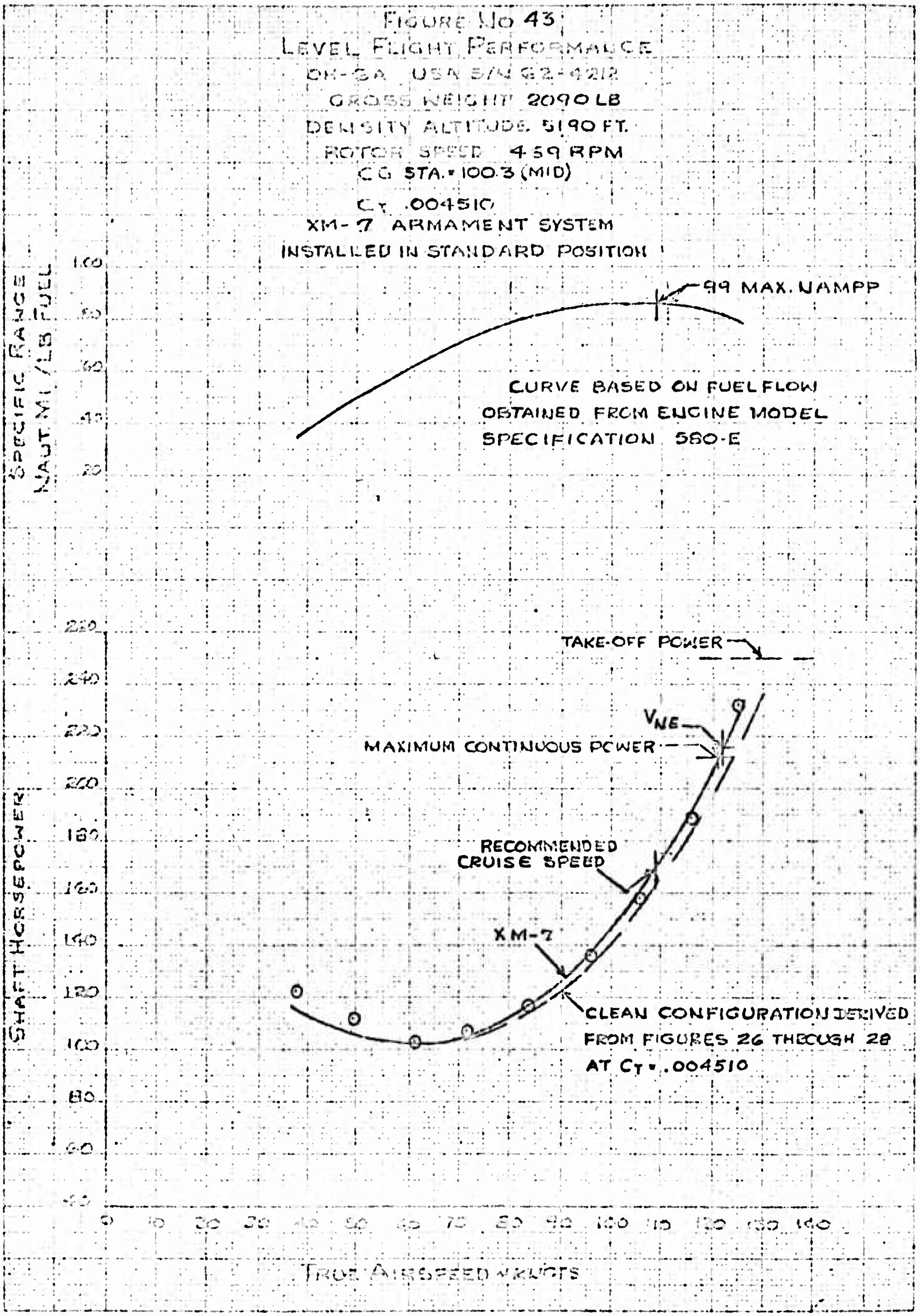


FIGURE NO 43
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2090 LB
DENSITY ALTITUDE 5190 FT.
ROTOR SPEED 459 RPM
CG STA. = 100.3 (MID)
C_T .004510
XM-7 ARMAMENT SYSTEM
INSTALLED IN STANDARD POSITION

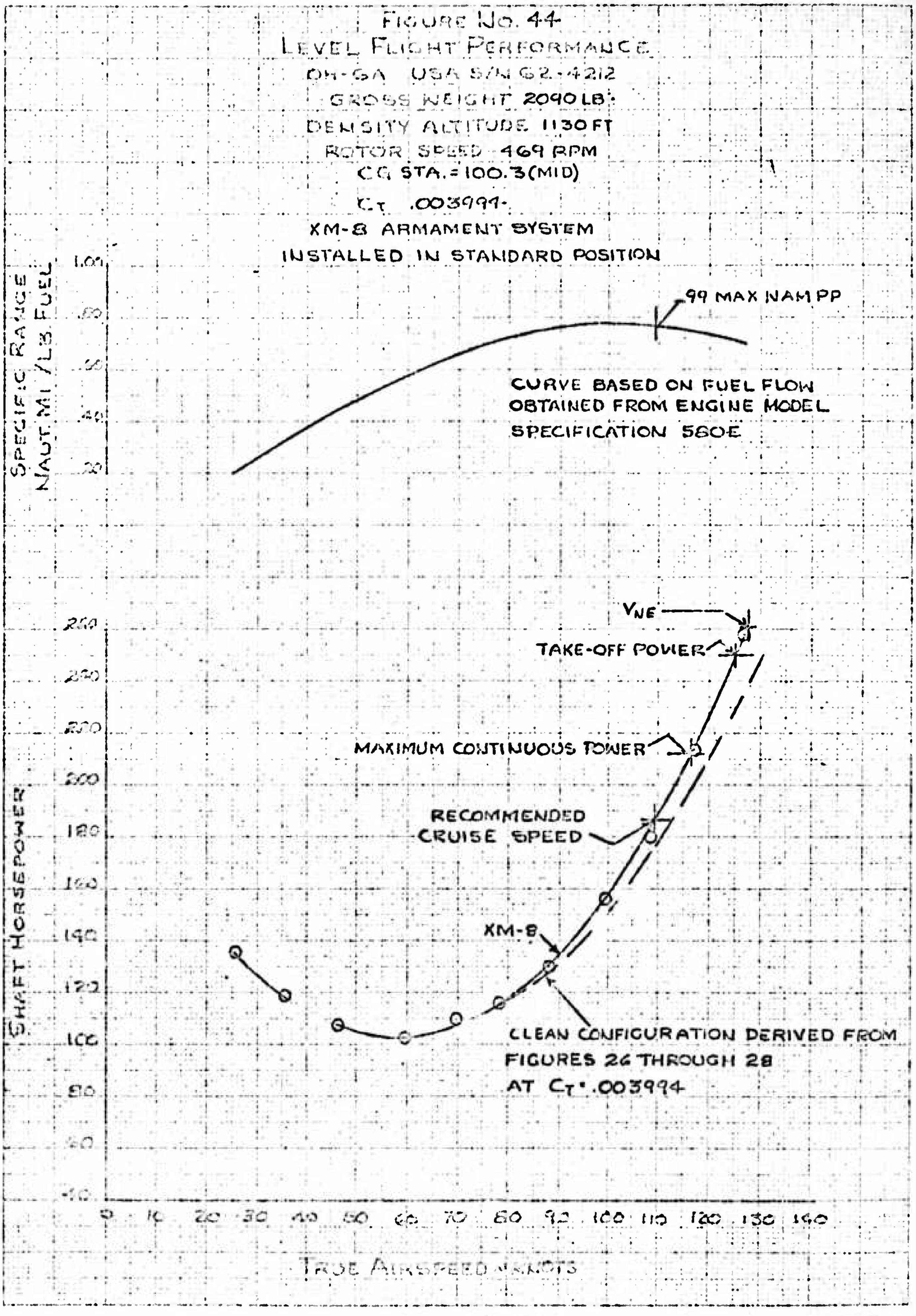


FIGURE No. 44
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2090 LB
DENSITY ALTITUDE 1130 FT
ROTOR SPEED 469 RPM
CG STA. = 100.3 (MID)
C_T .003994
XM-8 ARMAMENT SYSTEM
INSTALLED IN STANDARD POSITION
SPECIFIC RANGE
NAUT. MI / LB FUEL
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW
OBTAINED FROM ENGINE MODEL
SPECIFICATION 580E
V_{NE}
TAKE-OFF POWER
MAXIMUM CONTINUOUS POWER
RECOMMENDED
CRUISE SPEED
XM-8
CLEAN CONFIGURATION DERIVED FROM
FIGURES 26 THROUGH 28
AT C_T = .003994
SHAFT HORSEPOWER
TRUE AIRSPEED ~ KNOTS

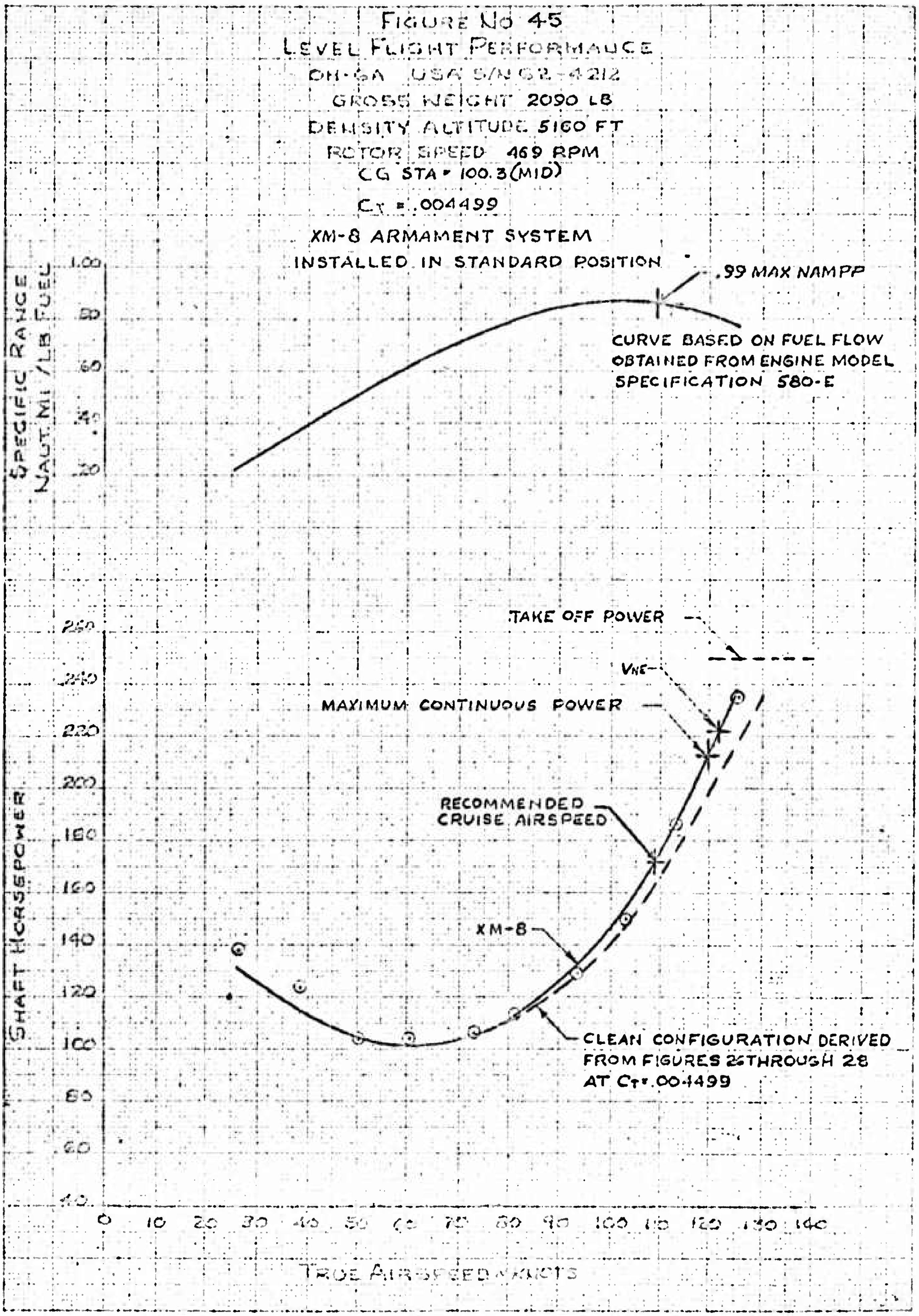


FIGURE No 45
LEVEL FLIGHT PERFORMANCE
OH-6A USA S/N 62-4212
GROSS WEIGHT 2090 LB
DENSITY ALTITUDE 5160 FT
ROTOR SPEED 469 RPM
CG STA = 100.3 (MID)
CT = .004499
XM-8 ARMAMENT SYSTEM
INSTALLED IN STANDARD POSITION
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
SPECIFIC RANGE NAUT. MI / LB FUEL
1.00
.80
.60
.40
.20
TAKE OFF POWER
VNE
MAXIMUM CONTINUOUS POWER
RECOMMENDED CRUISE AIRSPEED
XM-8
CLEAN CONFIGURATION DERIVED FROM FIGURES 26 THROUGH 28 AT CT = .004499
SHAFT HORSEPOWER
260
240
220
200
180
160
140
120
100
80
60
40
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
TRUE AIRSPEED ~ KNOTS

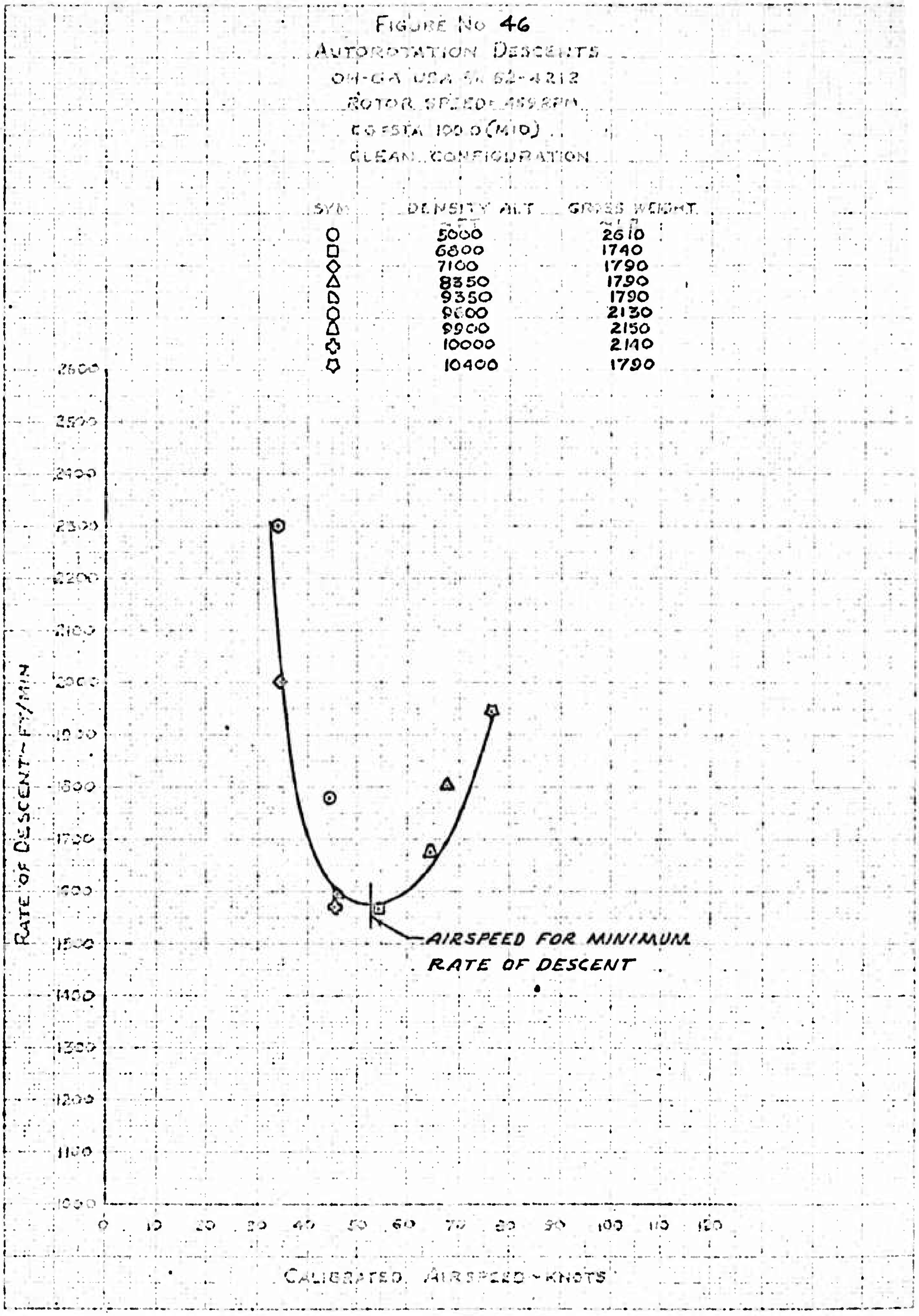


FIGURE No 46
AUTOROTATION DESCENTS
OH-6A USA S/N 62-4212
ROTOR SPEED = 459 RPM
C.G. STA 100.0 (MID)
CLEAN CONFIGURATION
SYM
DENSITY ALT
FT
GROSS WEIGHT
LB
5000
2610
6800
1740
7100
1790
8350
1790
9350
1790
9600
2130
9900
2150
10000
2140
10400
1790
AIRSPEED FOR MINIMUM
RATE OF DESCENT
RATE OF DESCENT ~ FT/MIN
CALIBRATED AIRSPEED ~ KNOTS

FIGURE NO. 47

AUTOROTATION DESCENTS

OH-6A USA S/N 62-4212

AIRSPEED 53 KNOTS CAS

CG = STATION NO 105 (MID)

CLEAN CONFIGURATION

SYM	DENSITY ALT ~ FT.	GROSS WEIGHT ~ LB
○	1900	2110
□	4850	2580
◇	6600	2080
△	9400	1790
D	9500	1890

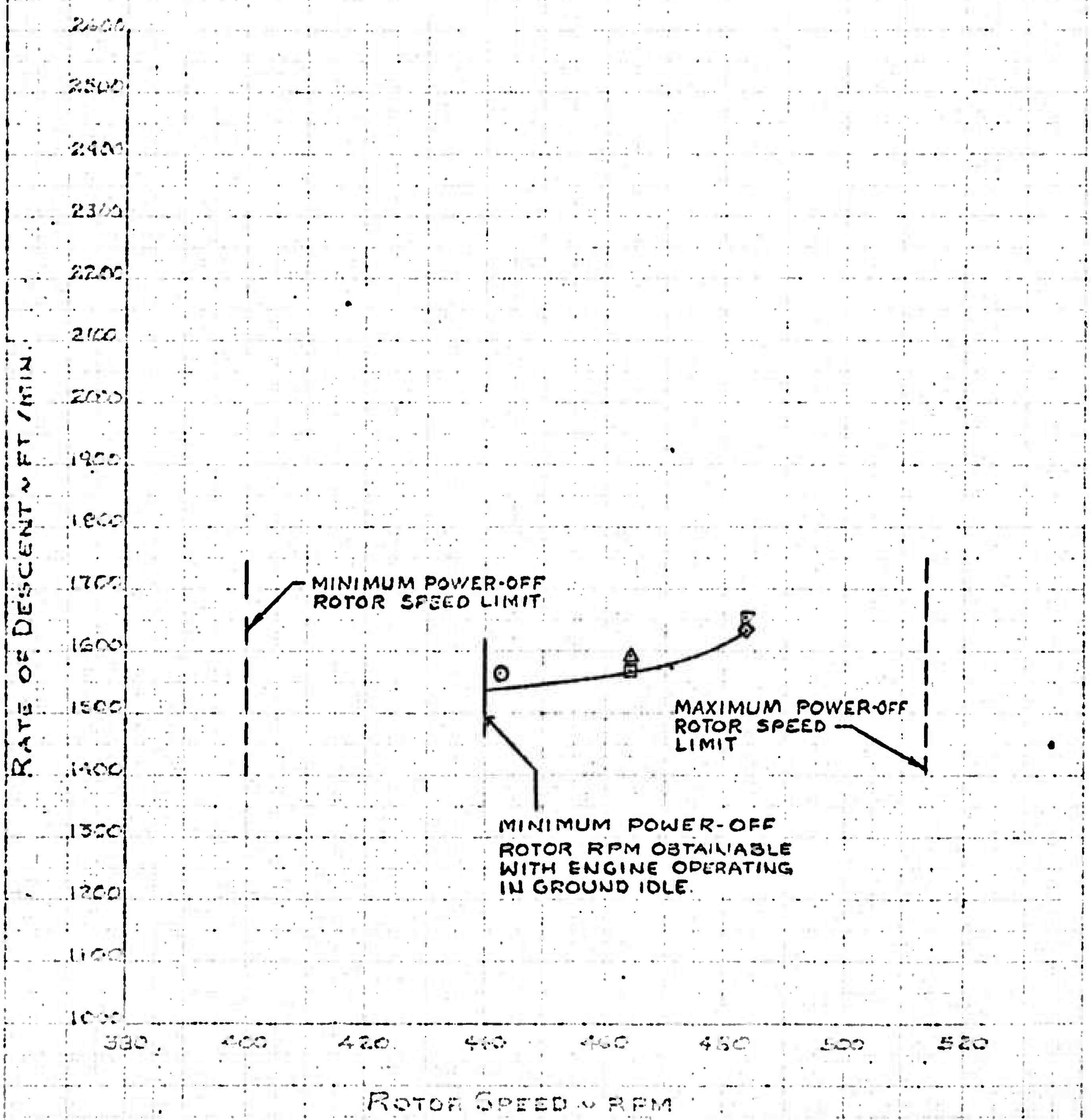


FIGURE NO. 48
AUTOROTATION DESCENTS
OH-6A USA S/N 62-4212
ROTOR SPEED=469 RPM
GROSS WEIGHT= 2085 LB
ALTITUDE = 5000 FT.
CG = STATION NO. 100 (MID)
CLEAN CONFIGURATION

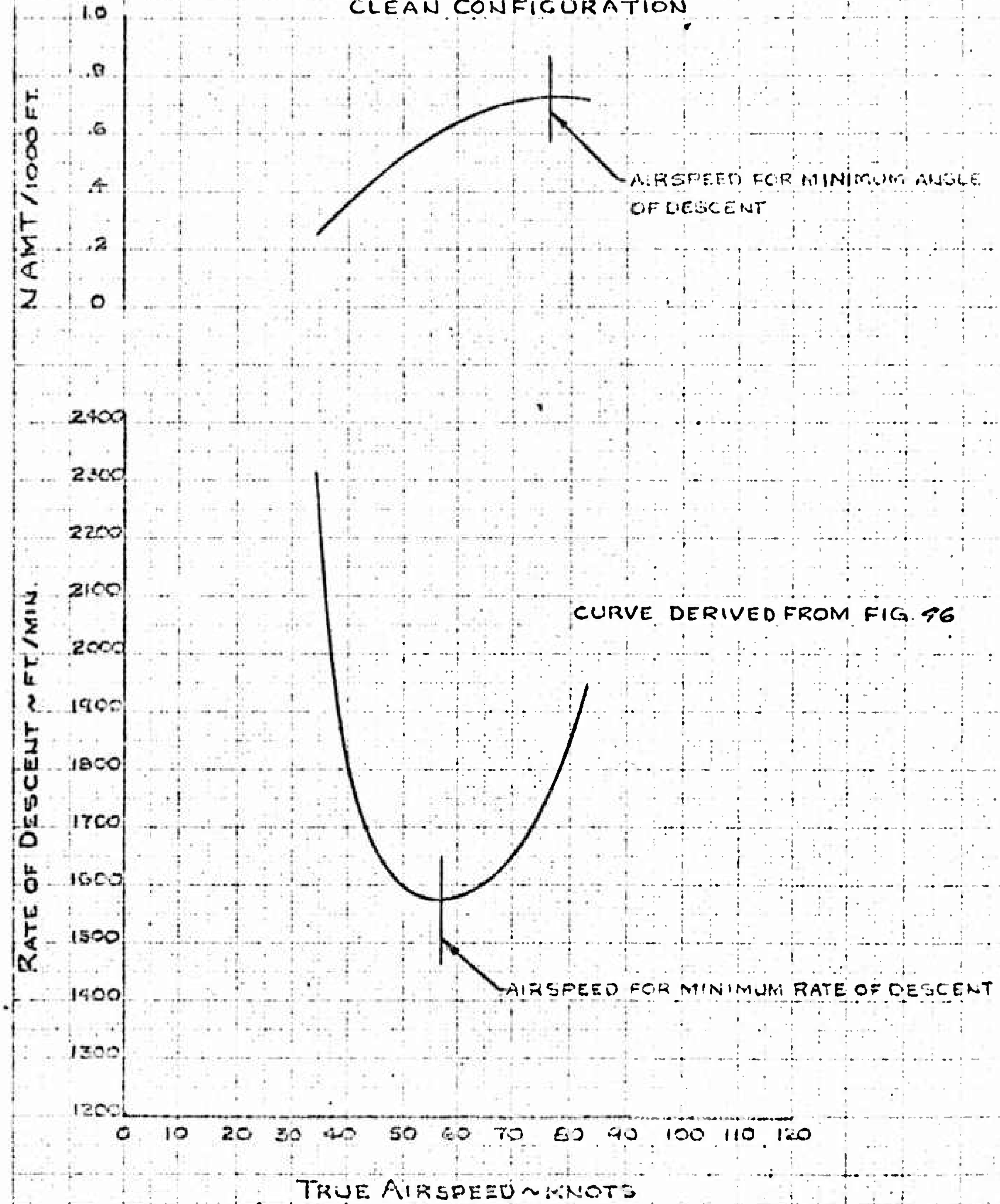


FIGURE No 49

SHAFT HORSEPOWER AVAILABLE

OH-6A USA S/N 62-4212

T-63-A-5

TAKE-OFF POWER

T_{T5} = 738°C

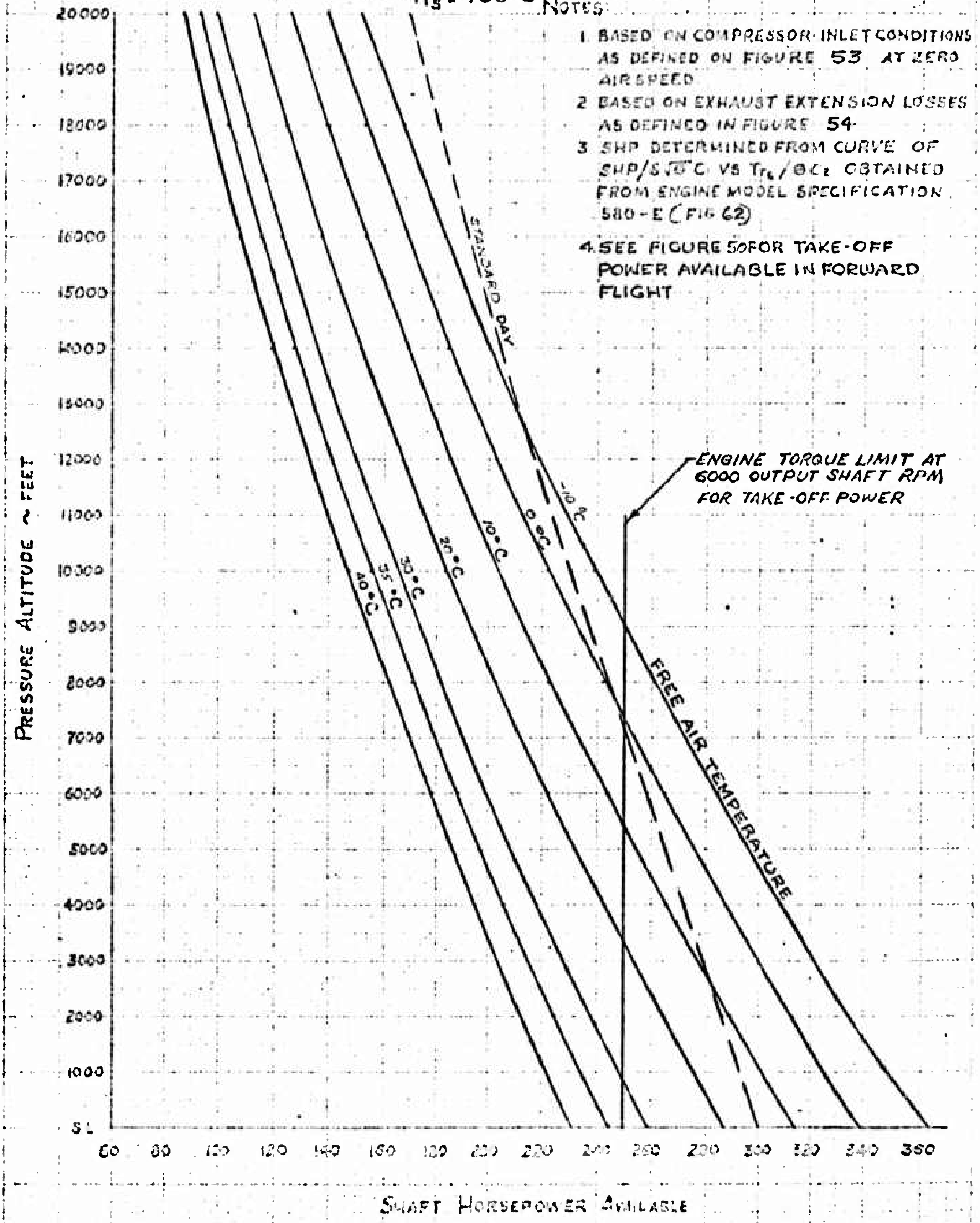


FIGURE No 50
SHAFT HORSEPOWER AVAILABLE
OH-6A USA S/N 62-4212

T63-A-5

TAKE-OFF POWER

US STANDARD DAY

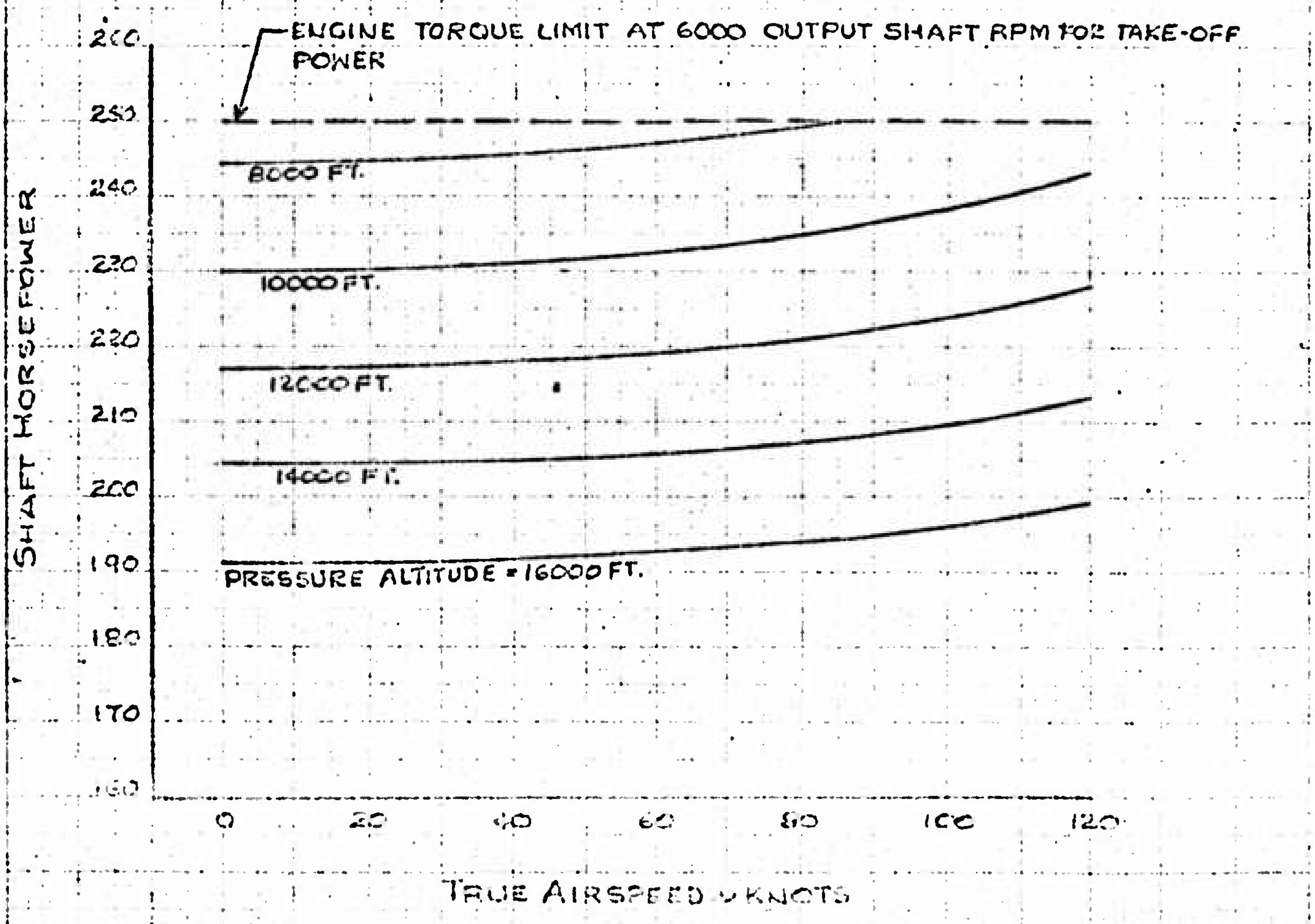
NOTES:

1. BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED ON FIGURE 53

2. BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 54

3. SHP DETERMINED FROM CURVE OF SHP/$\delta\sqrt{\theta}$C$_1$ VS $T_{T5}/\theta C_2$ AS OBTAINED FROM ENGINE MODEL SPECIFICATION 880E (FIGURE 62)

4. *SEE FIGURE 49 FOR TAKE OFF POWER AVAILABLE IN HOVER*



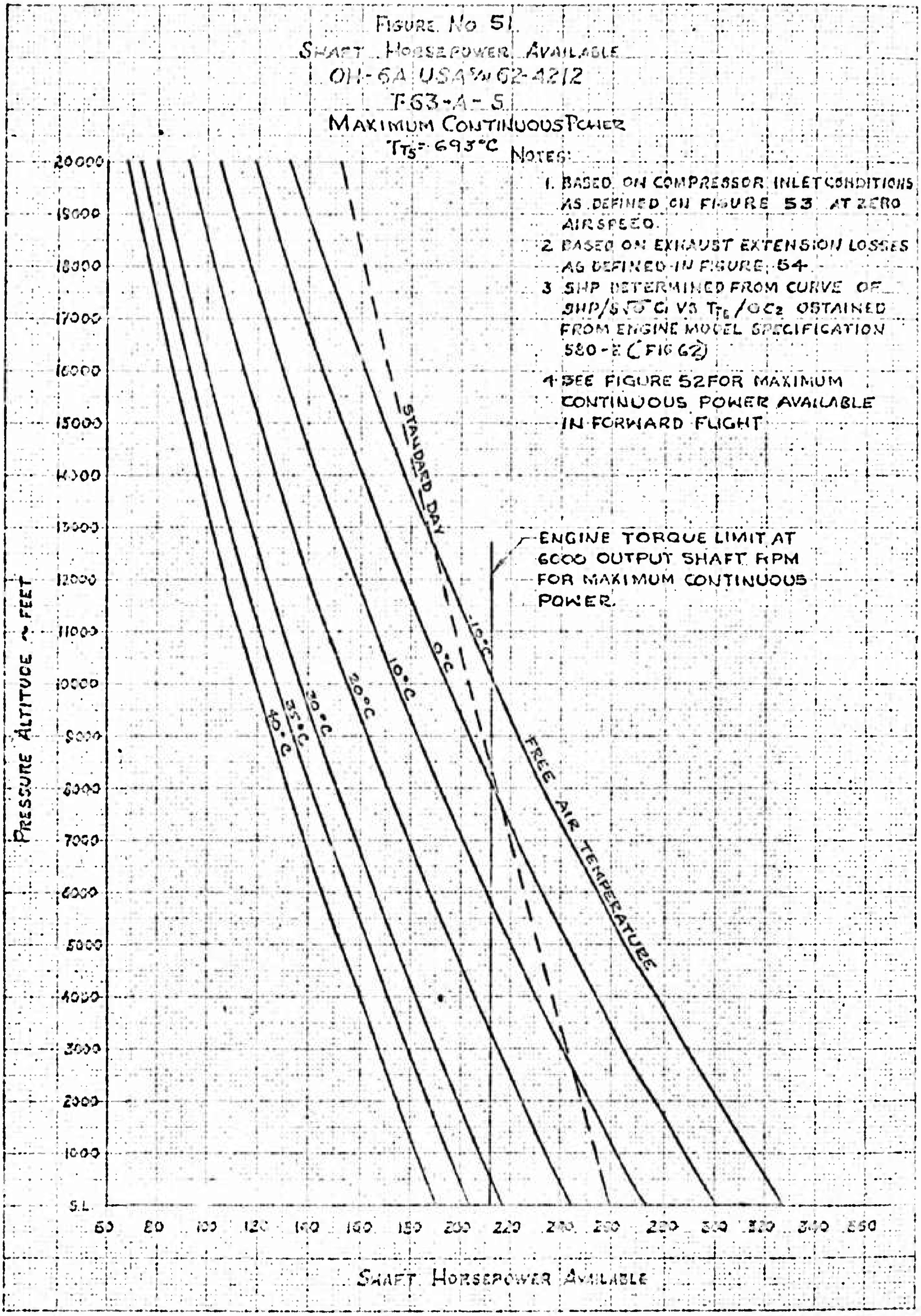


FIGURE No 51
SHAFT HORSEPOWER AVAILABLE
OH-6A USA S/N 62-4212
T63-A-5
MAXIMUM CONTINUOUS POWER
T_{T5} = 693°C
NOTES:
1. BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED ON FIGURE 53 AT ZERO AIRSPEED.
2. BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 54.
3. SHP DETERMINED FROM CURVE OF SHP/δ√θ C₁ VS T_{T5}/θC₂ OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E (FIG 62)
4. SEE FIGURE 52 FOR MAXIMUM CONTINUOUS POWER AVAILABLE IN FORWARD FLIGHT
ENGINE TORQUE LIMIT AT 6000 OUTPUT SHAFT RPM FOR MAXIMUM CONTINUOUS POWER.
STANDARD DAY
FREE AIR TEMPERATURE
-10°C
0°C
10°C
20°C
30°C
35°C
40°C
PRESSURE ALTITUDE ~ FEET
20000
19000
18000
17000
16000
15000
14000
13000
12000
11000
10000
9000
8000
7000
6000
5000
4000
3000
2000
1000
S.L.
60
80
100
120
140
160
180
200
220
240
260
280
300
320
340
360
SHAFT HORSEPOWER AVAILABLE

FIGURE No. 52
SHAFT HORSEPOWER AVAILABLE
OH-6A USA S/N 62-4212
T63-A-5
MAXIMUM CONTINUOUS POWER
US STANDARD DAY

NOTES:

1. BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED ON FIGURE 53
2. BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 54
3. SHP DETERMINED FROM CURVE OF $SHP/\delta_{C_1}\sqrt{\theta_{C_1}}$ VS T_{T_5}/θ_{C_2} AS OBTAINED FROM ENGINE MODEL SPECIFICATION 580E (FIGURE 62)
4. *SEE FIGURE 57 FOR MAXIMUM CONT. POWER AVAILABLE IN HOVER*

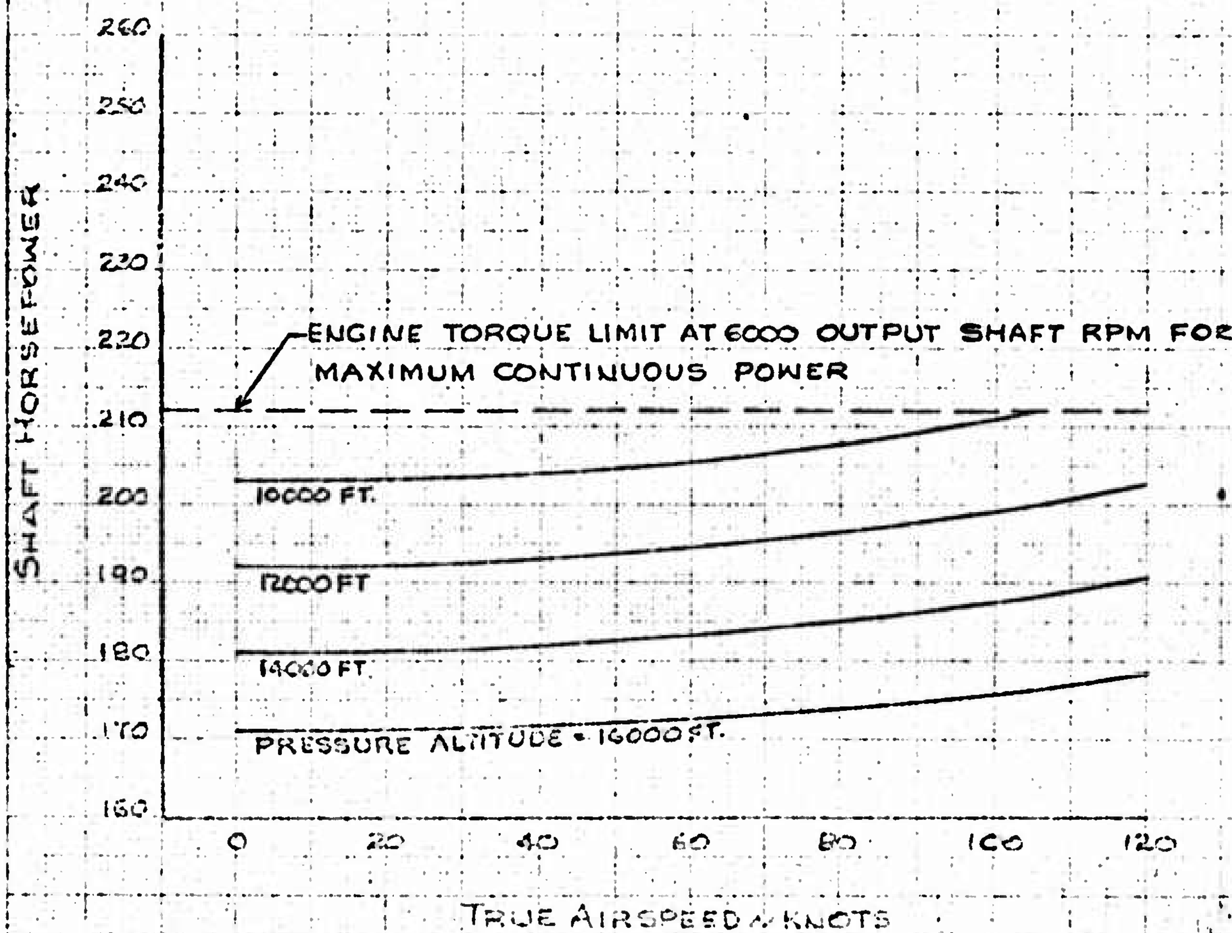


FIGURE NO. 53
ENGINE CHARACTERISTICS
OH-6A USA S/N 62-4212
T-63-A-5 S/N 400043

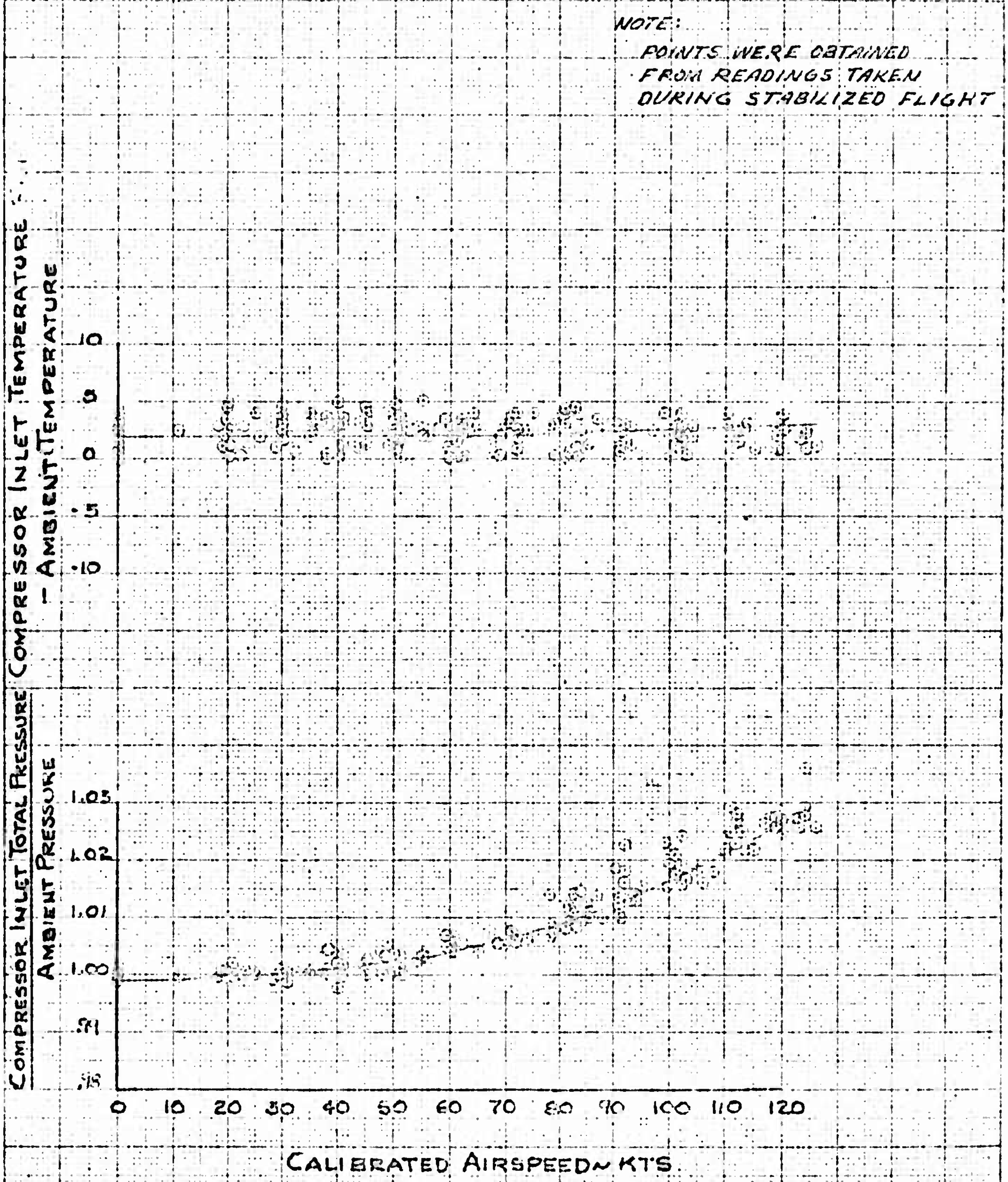
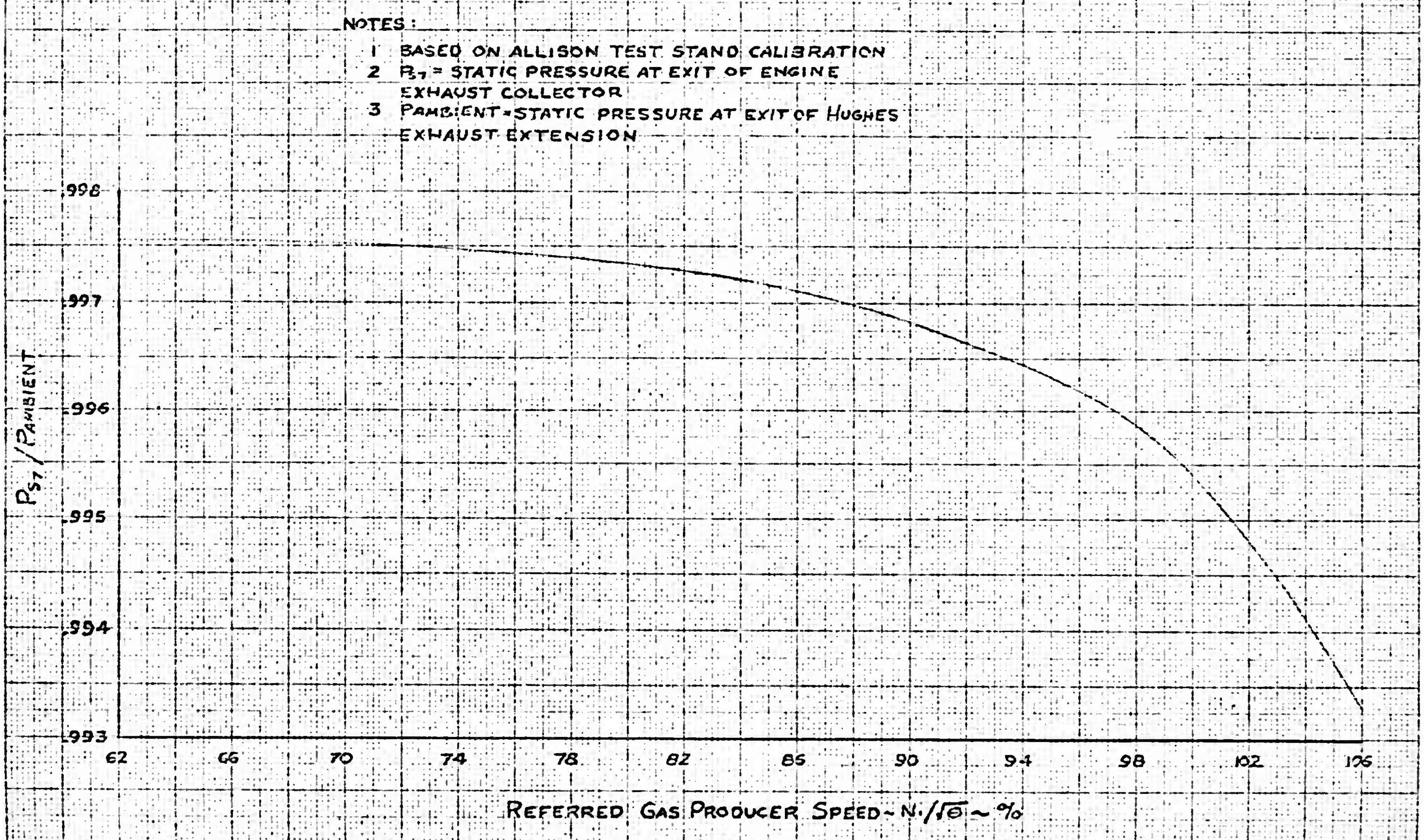


Figure No 54
Effect of Hughes Exhaust Extension
on T63-A-5 Exhaust Static Pressure



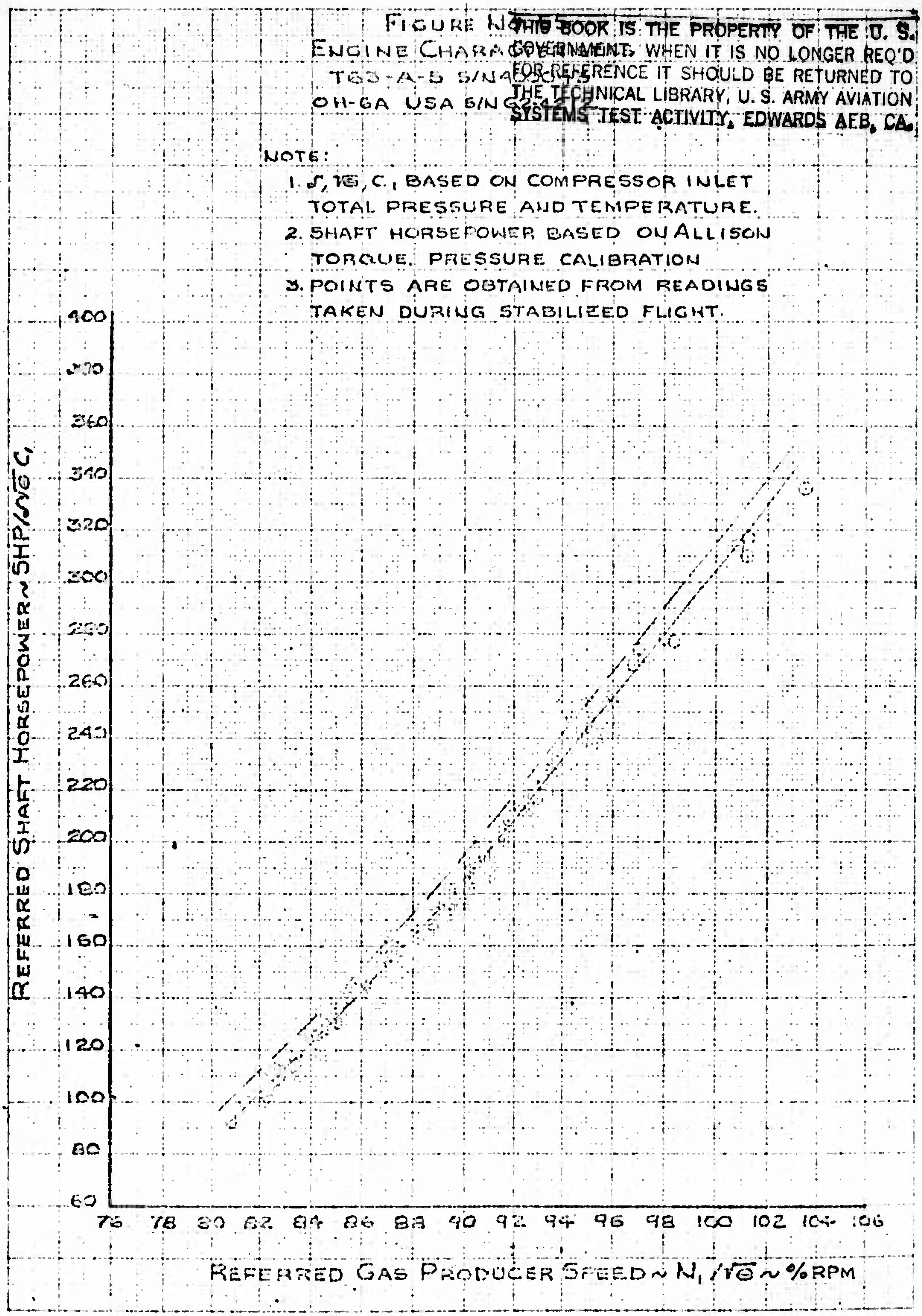
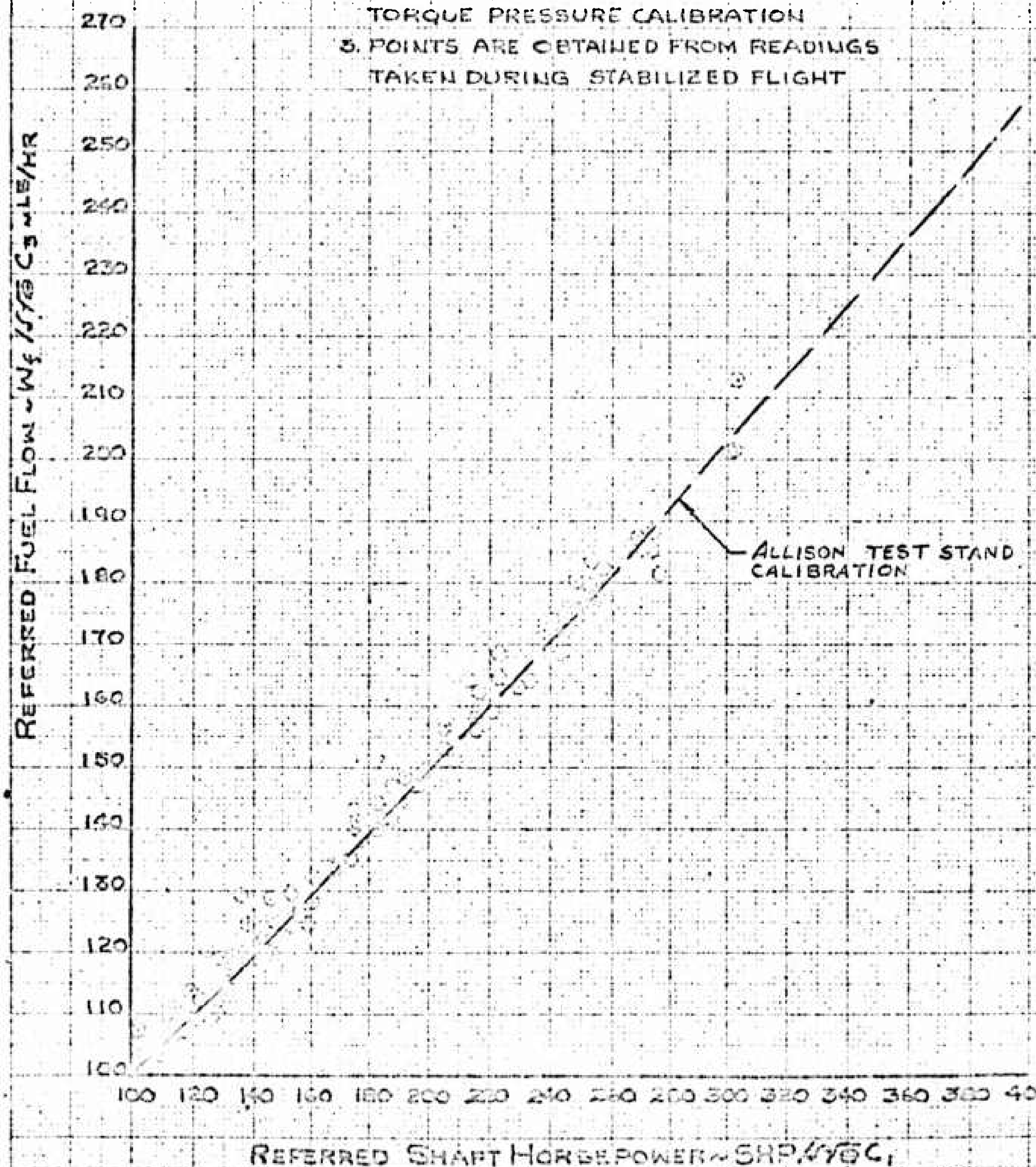


FIGURE N
ENGINE CHARA
T63-A-5 S/N4
OH-6A USA S/N
NOTE:
1. δ, √θ, C₁ BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE.
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION
3. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT.
REFERRED SHAFT HORSEPOWER ~ SHP/δ√θ C₁
400
360
340
320
300
260
240
220
200
180
160
140
120
100
80
60
76
78
80
82
84
86
88
90
92
94
96
98
100
102
104
106
REFERRED GAS PRODUCER SPEED ~ N₁/√θ ~ % RPM

FIGURE NO. 56
ENGINE CHARACTERISTICS
T63-A-5 S/N 400043
OH-6A USA S/N 62-4212

NOTE:
1. δ, $\sqrt{\theta}$, C_1 AND C_3 BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION
3. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT



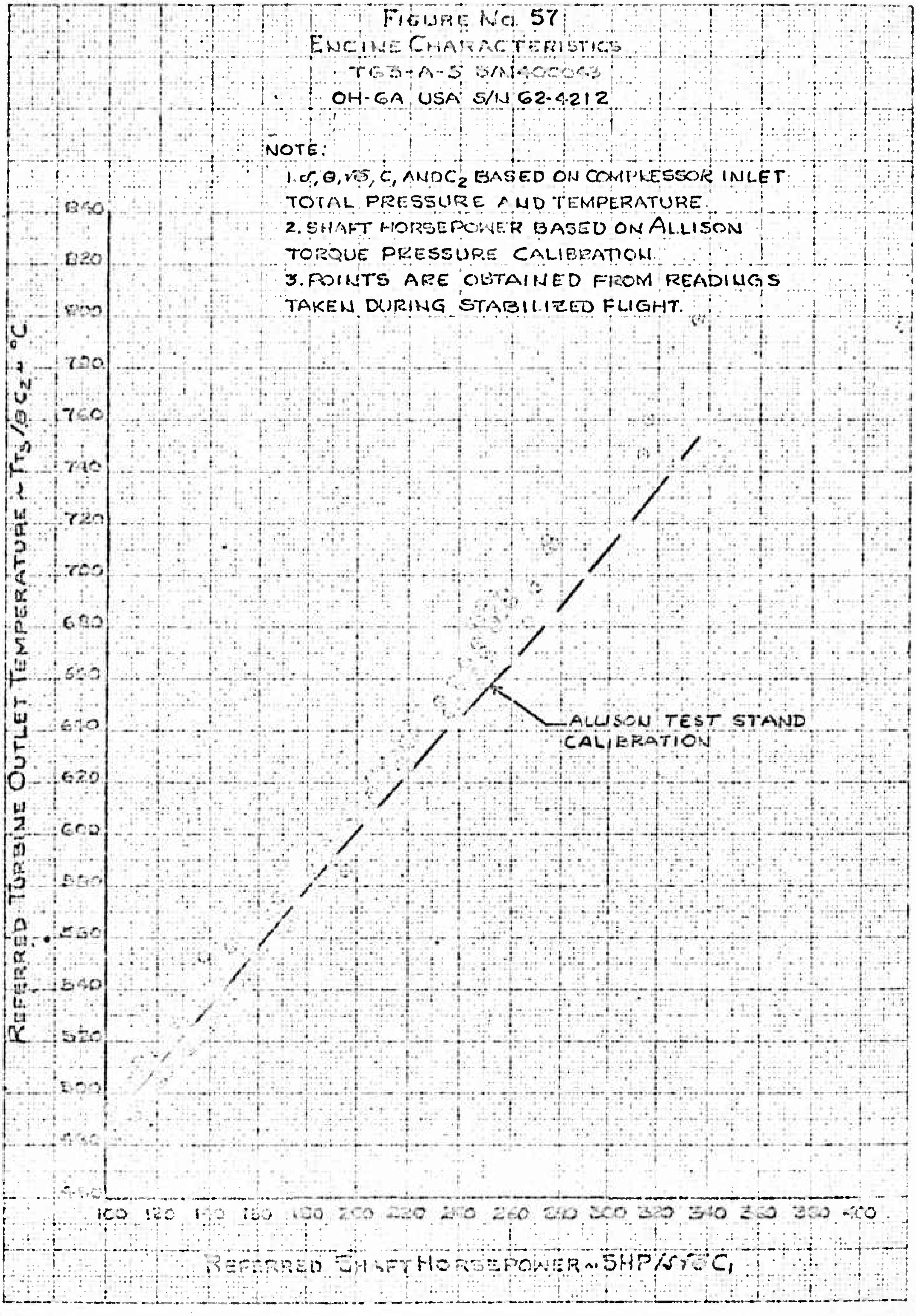


FIGURE NO. 57
ENGINE CHARACTERISTICS
T63-A-5
OH-6A USA S/N 62-4212
NOTE:
1. σ, θ, √θ, C, AND C2 BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION.
3. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT.
ALLISON TEST STAND CALIBRATION
REFERRED TURBINE OUTLET TEMPERATURE ~ TT5/θC2 ~ °C
840
820
760
740
720
700
680
620
600
540
520
REFERRED SHAFT HORSEPOWER ~ SHP/δ√θC1
100
120
200
220
260
300
320
340
400

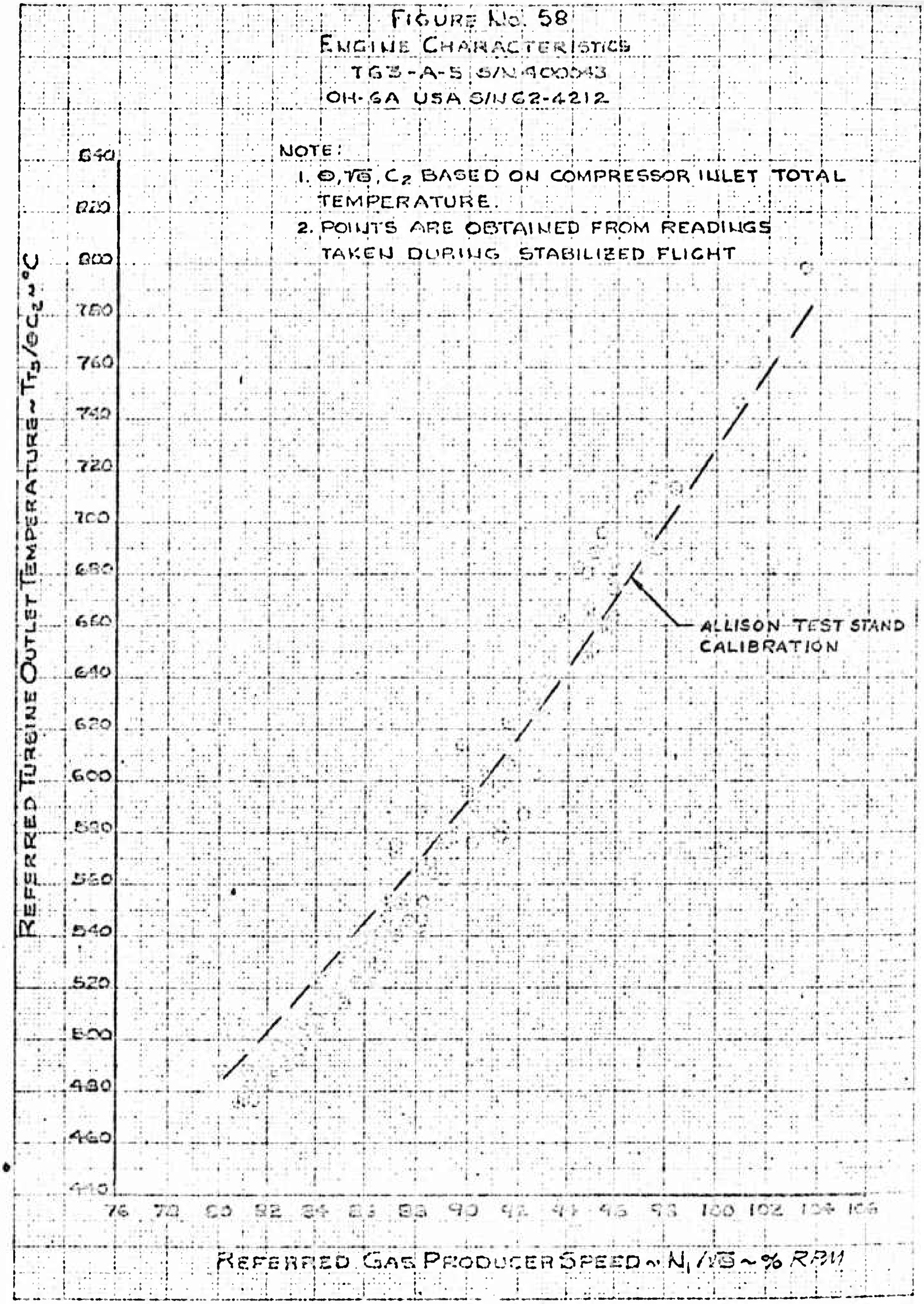
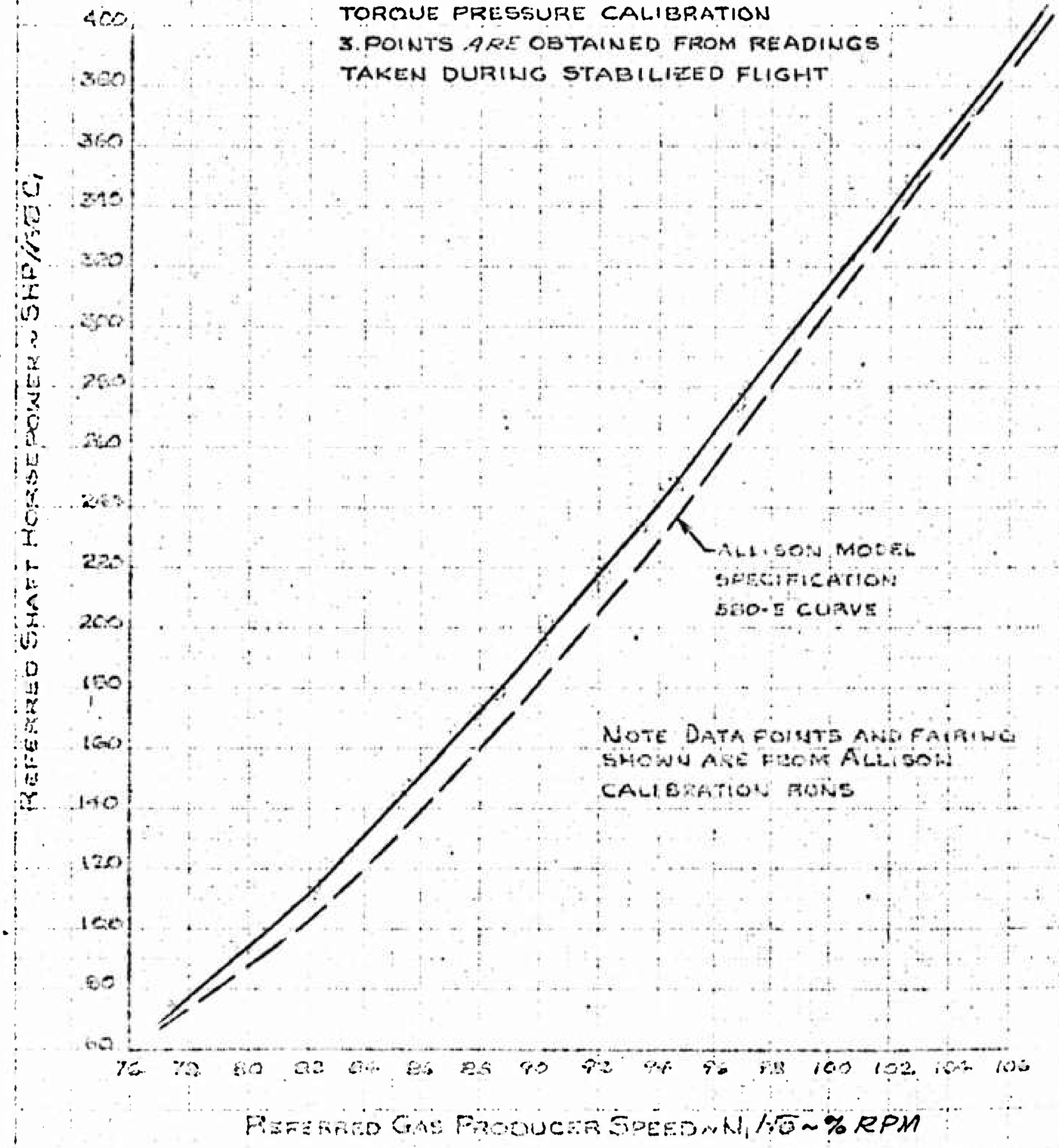


FIGURE No. 58
ENGINE CHARACTERISTICS
T63-A-5 S/N 400543
OH-6A USA S/N 62-4212
NOTE:
1. Θ, √Θ, C2 BASED ON COMPRESSOR INLET TOTAL TEMPERATURE.
2. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT
ALLISON TEST STAND CALIBRATION
REFERRED TURBINE OUTLET TEMPERATURE ~ TT5/ΘC2 ~ °C
840
820
800
780
760
740
720
700
680
660
640
620
600
580
560
540
520
500
480
460
440
76
78
80
82
84
86
88
90
92
94
96
98
100
102
104
106
REFERRED GAS PRODUCER SPEED ~ N1/√Θ ~ % RPM

FIGURE No. 59
ENGINE CHARACTERISTICS
T63-A-5 S/N 400043
BASED ON ALLISON TEST
STAND CALIBRATION

NOTE:
1. δ, $\sqrt{\theta}$, AND C_1 BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION
3. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT



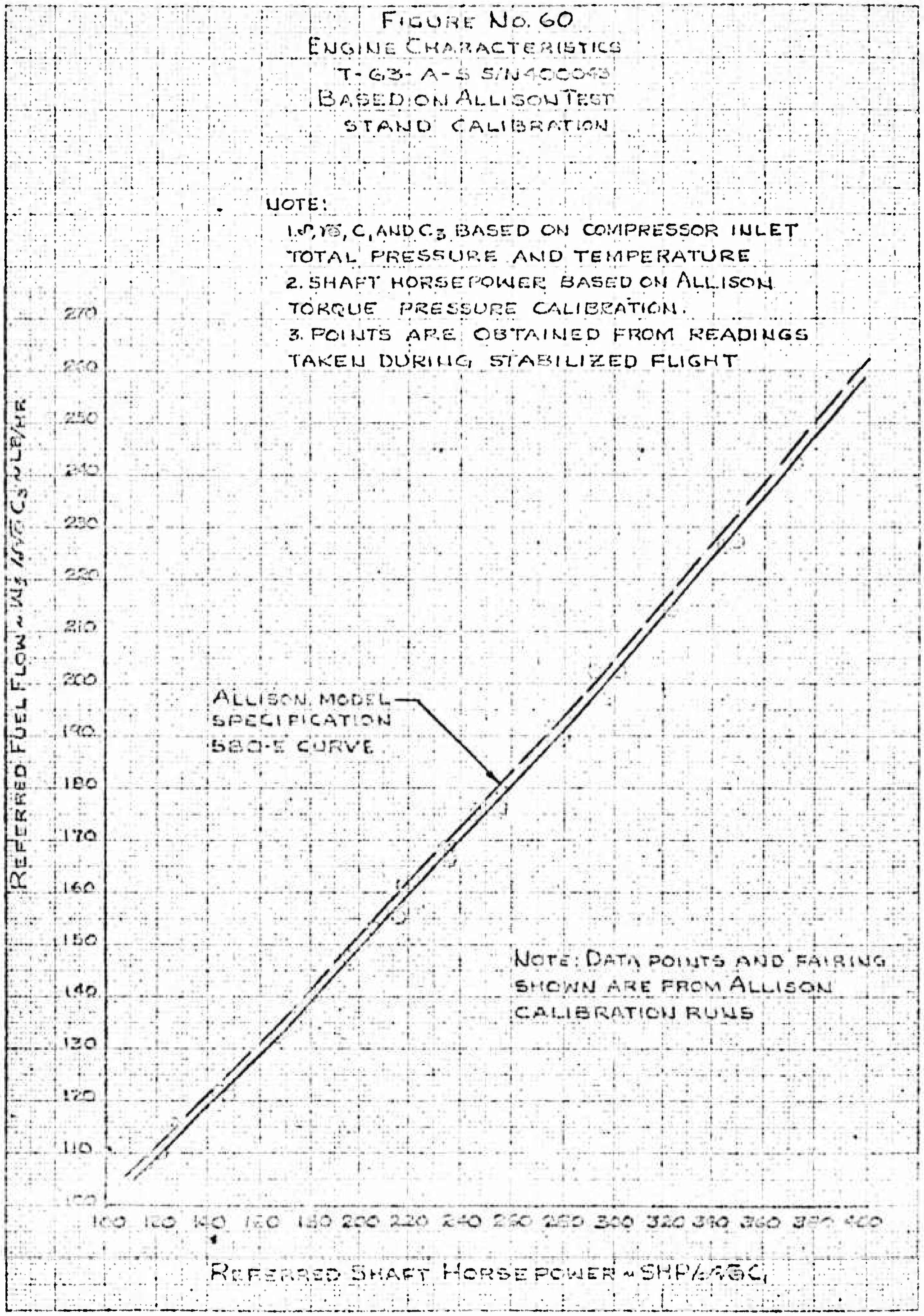


FIGURE No. 60
ENGINE CHARACTERISTICS
T-63-A-5 S/N400043
BASED ON ALLISON TEST
STAND CALIBRATION
NOTE:
1. P, T, C1 AND C3 BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION.
3. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT
ALLISON MODEL SPECIFICATION 580-E CURVE
NOTE: DATA POINTS AND FAIRING SHOWN ARE FROM ALLISON CALIBRATION RUNS
REFERRED FUEL FLOW ~ LB/HR
REFERRED SHAFT HORSEPOWER ~ SHP
100
110
120
130
140
150
160
170
180
190
200
210
220
230
240
250
260
270
120
140
180
200
220
240
260
300
320
340
360
400

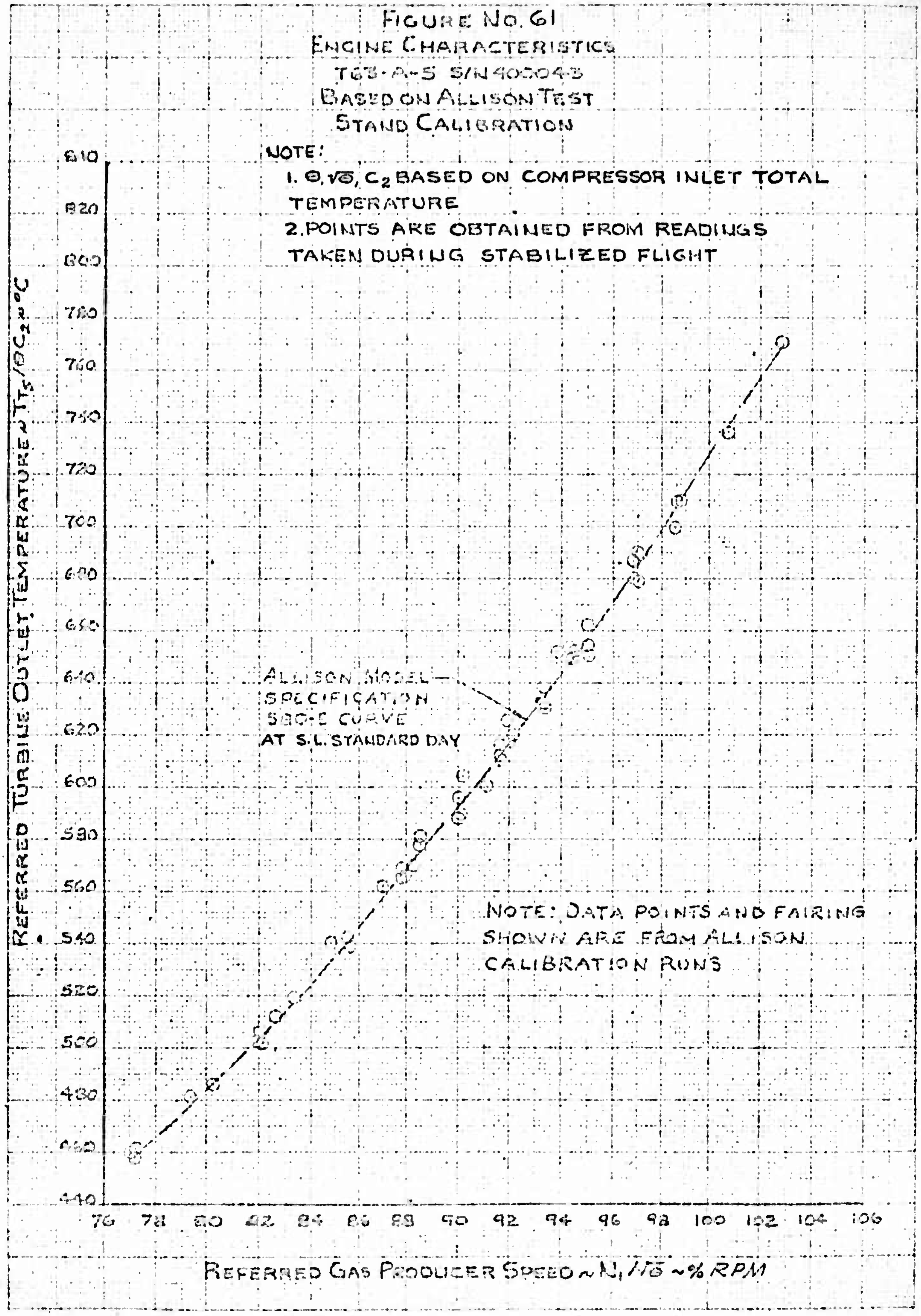


FIGURE NO. 61
ENGINE CHARACTERISTICS
T63-A-5 S/N 400043
BASED ON ALLISON TEST
STAND CALIBRATION
NOTE:
1. Θ, √Θ, C₂ BASED ON COMPRESSOR INLET TOTAL TEMPERATURE
2. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT
ALLISON MODEL SPECIFICATION 580-E CURVE AT S.L. STANDARD DAY
NOTE: DATA POINTS AND FAIRING SHOWN ARE FROM ALLISON CALIBRATION RUNS
REFERRED TURBINE OUTLET TEMPERATURE ~ TT5/ΘC2 ~ °C
REFERRED GAS PRODUCER SPEED ~ N1/√Θ ~ % RPM

FIGURE NO. 62
ENGINE CHARACTERISTICS
T63-A-5 S/N 400043
BASED ON ALLISON TEST
STAND CALIBRATION

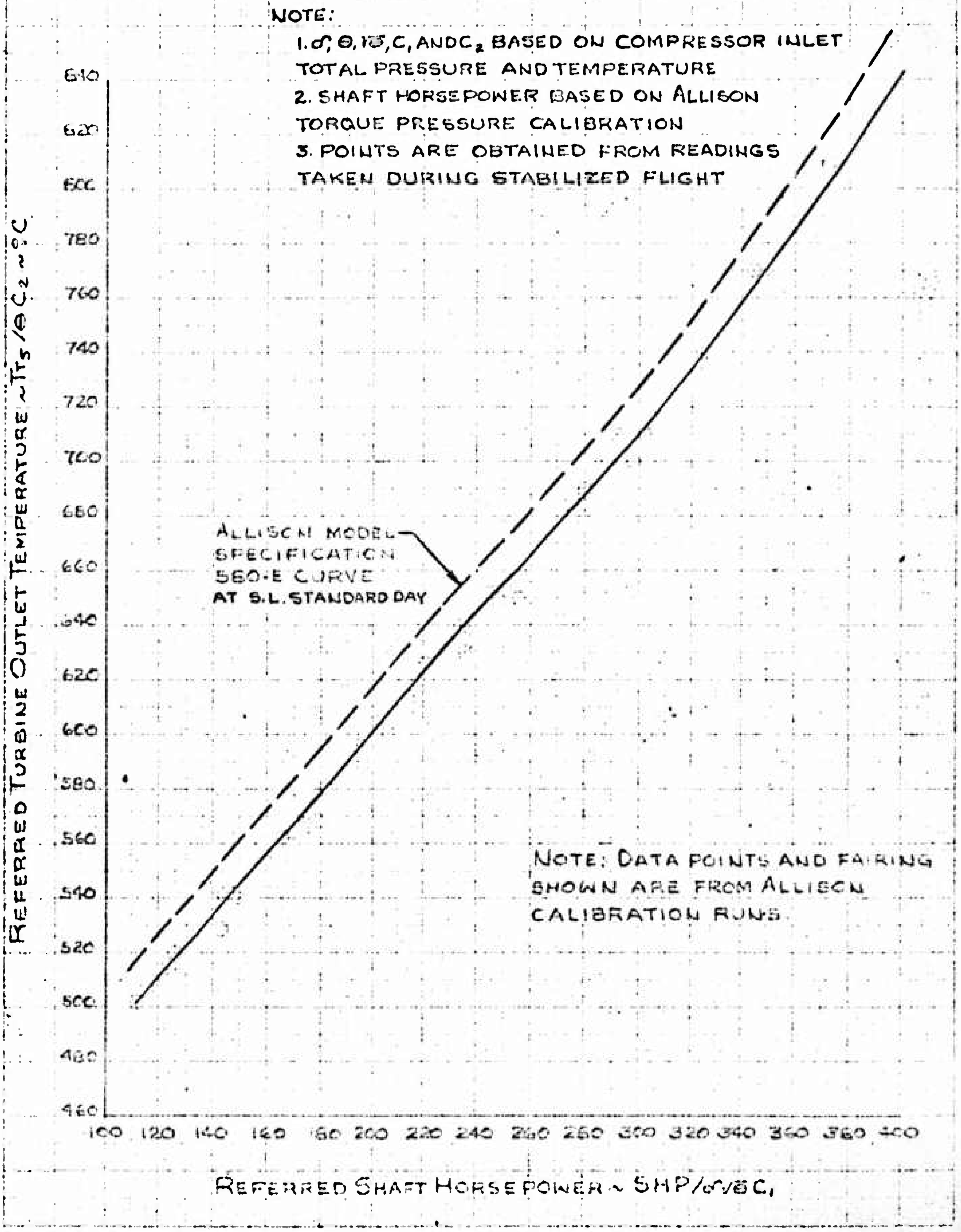


FIGURE NO 63
AIRSPEED CALIBRATION
OH-6A USA S/N 62-4212
BOOM SYSTEM
GROUND SPEED COURSE

AVG H_D ~FT.	AVG G.W ~LB	AVG C.G ~IN.	ROTOR SPEED ~RPM
1670	2090	101.3	469

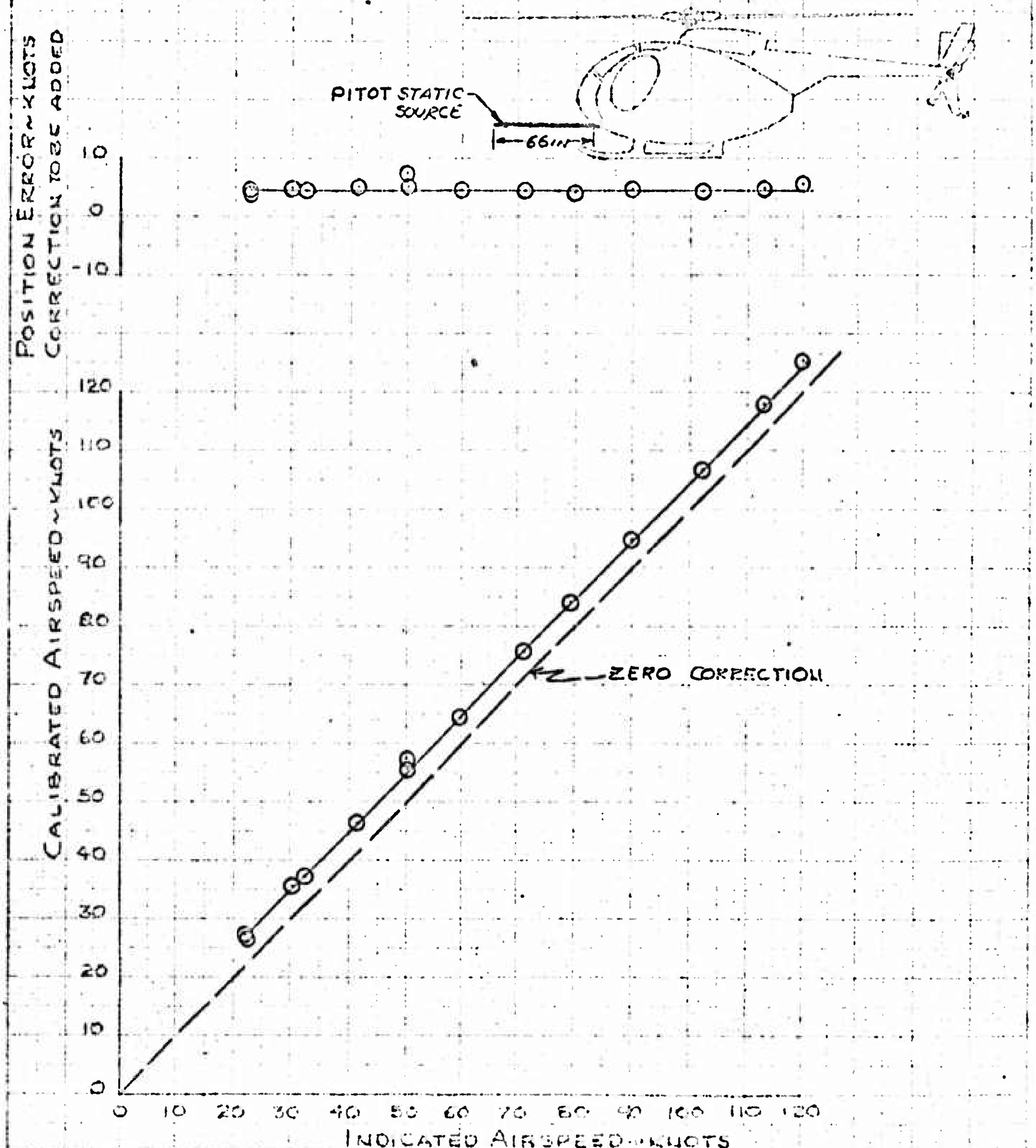


FIGURE NO 64

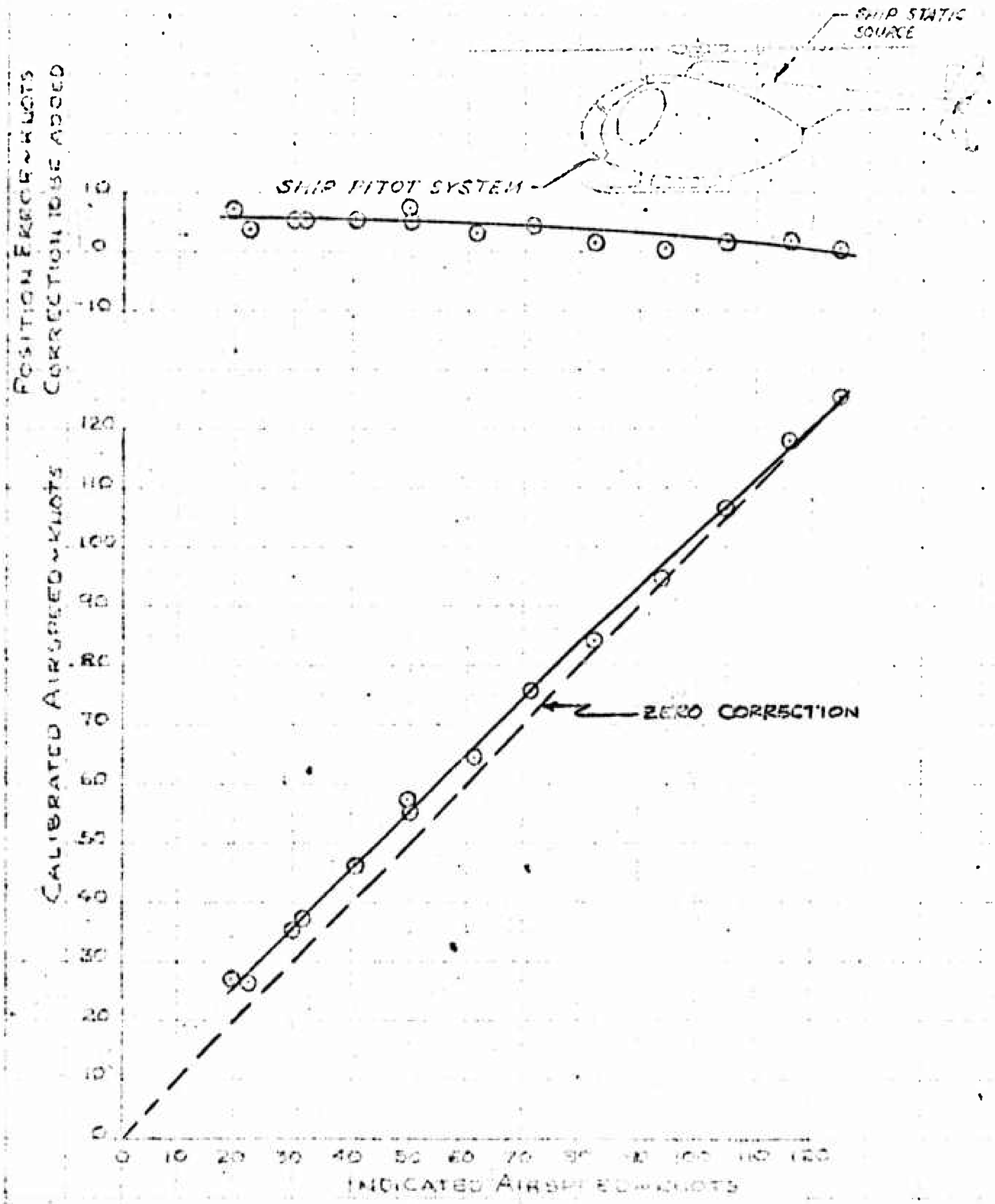
AIRSPEED CALIBRATION

OH-6A USA S/N 62-4212

STANDARD SYSTEM

GROUND SPEED COURSE

AVG H_D ~FT	AVG GW ~LB	AVG C.G. ~IN	ROTOR SPEED ~RPM
1670	2090	101.3	469



APPENDIX II

GENERAL AIRCRAFT INFORMATION

Aircraft Dimensions, Design Data, FAA Type Inspection and Authorization Limitations, Weight and Balance, Instrumentation, Engine Model Specification 580-E Inaccuracies, and Engine Operation

1. Sources of Information

See Part I, Appendix II

2. Description of Aircraft and Systems

2.1 Aircraft Design Data

a. Aircraft Dimensions and Certified Weights

See Part I, Appendix II

b. Rotor Blade Control Travel

See Part I, Appendix II

c. Rotor Dimensions and Design Data

See Part I, Appendix II

Additional Information:

Rotor Solidity	.054
Disc Loading (at design gross weight)	3.9

2.2 Aircraft Systems

2.2.1 Electrical Systems

See Part I, Appendix II

2.2.2 Power Plant

The T63-A-5 turbo-shaft engine has a nominal rating of 250 Shaft Horsepower (SHP). As installed in the OH-6A, the engine is limited by either the output shaft torque or the gas producer turbine outlet temperature (T_{t_5}). For maximum continuous operation, these limits were 186 foot-pounds torque at 6000 rpm (212 SHP) or 693 degrees Centigrade (C) T_{t_5}, whichever is reached first. For takeoff power (maximum of 5 minutes continuous operation), these limits are 219 foot-pounds torque (250 SHP) or 738 degrees C.

The engine is a free turbine type. The compressor consists of 6 axial stages and 1 centrifugal stage. Compressor speed at 100 percent is 51,120 rpm. The combustor section consists of a single chamber where a regulated flow of fuel is injected to support continuous combustion. The power turbine has 2 axial stages. Power turbine speed at 100 percent is 35,000 rpm. The high speed of the power turbine is reduced in the accessory gear box to 6000 rpm for the engine output speed. Engine operated accessories are also driven from the accessory gear box.

The DP-D3 gas turbine fuel control is pneumatically operated by compressor discharge air. The fuel control senses input from 3 sources. These sources are the pilot's twist grip, the fly-ball governor connected to the gas producer and the power turbine governor. In addition, both altitude and temperature compensation are provided. The function of the fuel control is to integrate the inputs so that the power turbine speed selected by the pilot is maintained under varying load demands.

A steady-state "droop" is built into the fuel control. This means that when the engine load is increased or decreased, the speed of the power turbine will change slightly. The "droop" is required to avoid rotor-engine dynamic instability and "hunting" during steady-state operation. In an attempt to eliminate some of the undesirable effects of the"droop", a "droop" compensator cam is installed in the linkage between the collective pitch control and the power turbine governor. This cam converts collective control movement to an input to the power turbine governor which anticipates the changing engine load. The cam was designed to reduce transient "droop" and to eliminate steady-state "droop" during rapid collective pitch applications. Operation of this "droop" compensator, as installed, was satisfactory throughout the test program.

There is no provision for emergency control in case of a fuel control failure. In the case of an uncontrolled overspeed in the power turbine, rotor speed and engine power may be controlled through pilot coordination of collective pitch control and twist grip rotation as in an ungoverned reciprocating engine.

2.2.3 Landing Gear System

See Part I, Appendix II

2.2.4 Fuel System

See Part I, Appendix II

2.2.5 Control System

See Part I, Appendix II

3. TIA Limitations

See Part I, Appendix II

4. Weight and Balance

The test aircraft was weighed prior to installation of test instrumentation. The weighing was done in a closed hangar using an electronic weighing kit. As weighed, the aircraft gross weight was 1069 pounds with the longitudinal C.G. located at Station 112.7.

In order to provide a basis for verifying whether the manufacturer's helicopter met the performance guarantees, the empty weight was defined to be the weight of OH-6A, USA S/N 62-4212, as delivered to the U. S. Army Aviation Test Activity, with corrections for the increase in weight of Government furnished equipment, and corrections for items installed but not part of the contract empty weight (See Part I, Section 1, paragraph 1.1.m). With these corrections, the empty weight used was 1101.5 pounds (full oil and no fuel). Full oil capacity is 11 pounds.

The manufacturer's performance guarantees were at a specified normal gross weight. This weight is defined as:

Normal Gross Weight	=	Empty weight + useful load	
Empty Weight	=		1101.5 lb
Useful Load	=	Pilot	200 lb
		Cargo	400 lb
		Fuel as necessary to meet normal gross weight of Model Specification, paragraph 3.2.1--383.5 lb	

OH-6A Model Specification, paragraph 3.2.1 states that the normal gross weight (performance guarantee weight) shall not exceed 2050 pounds ± 5 percent; therefore, helicopter performance was calculated at 2085 pounds (2050 + 1.7 percent).

Weights of some items not included in the empty weight which may be required for various missions are as follows:

Copilot flight controls	-	8.7 lb
Anti-collision light	-	1.0 lb
Ground handling wheels	-	49 lb
XM-7 armament system (full ammunition)	-	226 lb
XM-8 armament system (full ammunition)	-	257 lb

After installation of test instrumentation, the helicopter was again weighed. The basic weight (full oil and no fuel) was 1416 pounds with the longitudinal C.G. location at Station 109.3.

5. <u>Test Instrumentation</u>

The test instrumentation used during this evaluation was supplied, installed and maintained by the Logisitics Division of the U. S. Army Aviation Test Activity. Calibration of the instrumentation was accomplished by the Instrumentation Branch of the Air Force Flight Test Center; California Division Lockheed Aircraft Corporation; Douglas Aircraft Missile and Space Division; and the Logisitics Division of the U. S. Army Aviation Test Activity.

A swivel mounted pitot-static airspeed head was installed on a nose boom mounted approximately 5 feet forward of the nose of the helicopter. The static pressure ports of this pitot-static head were the pressure source for the sensitive altimeter as well as the sensitive boom airspeed indicator. The airspeed position error for this installation is shown in Figure No. 63, Section 3, Appendix I. Sensitive instrumentation was installed prior to initiation of the test flights to measure the following parameters:

<u>Pilot-Engineer Panel:</u>

Boom System Airspeed

Standard System Airspeed

Boom Altitude

Rate of Climb

Angle of Sideslip

Free Air Temperature

Rotor Speed

Gas Producer Speed (N_1)

Torquemeter Oil Pressure

Turbine Outlet Temperature (T_{t5})

Compressor Inlet Total Temperature

Compressor Inlet Total Pressure

Exhaust Gas Static Pressure

Cockpit Absolute Pressure

Total Fuel Used

Photo Panel Frame Counter

Oscillograph Record Counter

<u>Photo Panel:</u>

Boom Altitude

Time of Day

Free Air Temperature

Gas Producer Speed (N_1)

Torquemeter Oil Pressure

Compressor Inlet Total Pressure

Compressor Discharge Total Pressure

Combustion Static Pressure

Compressor Discharge Temperature

Fuel Control Inlet Fuel Temperature

Total Fuel Used

Photo Panel Frame Counter

Oscillograph Record Counter

Recording Oscillograph:

Gas Producer Speed

Rotor Speed

Power Turbine Speed

Gas Producer Control Lever Position

C.G. Vertical Acceleration (vibration)

C.G. Lateral Acceleration (vibration)

Pilot's Station Vertical Acceleration (vibration)

Pilot's Station Lateral Acceleration (vibration)

Pilot's Event

Engineer's Event

Bridge-Balance Voltage

6. Engine Model Specification 580-E Inaccuracies

During the course of the performance tests, inaccuracies were encountered while working with the T63-A-5 Engine Model Specification 580-E. The following information is presented to clarify this situation.

The T63-A-5 Engine Model Specification 580-E contains engine performance data that do not represent an actual engine. These data are based primarily on an uninstalled power available of 206 Shaft Horsepower (SHP) at 6000 feet pressure altitude, 95 degree Fahrenheit (F) ambient temperature and a sea level standard day specific fuel consumption of .71 pounds of fuel used per hour per SHP at 250 SHP. The sea level condition was a guarantee point and the SHP available at 6000 feet, 95 degrees F, was a power condition contained in an earlier model specification (580-A).

In order to construct a model specification for a new engine, tests on various engine components (i.e., compressor, combustor, turbines, etc.) are conducted by the manufacturer. From these tests, the engine performance is predicted for various altitudes and ambient temperatures. As completed engines are run and more information is obtained, the predicted performance is revised.

When the engines were calibrated for the LOH flight test programs, it became apparent that the power deterioration with increasing altitude and temperature was greater than predicted. A specification engine that would just meet the performance guarantee at sea level on a standard day would not produce 206 SHP uninstalled at 6000 feet and 95 degrees F. Such an engine would only produce 199 SHP at 6000 feet, 95 degrees F.

The engine calibration test data showed that the T63-A-5 engine had a performance margin over the 580-E specification values at sea level. The existence of this sea level power margin allowed 206 SHP to be obtained at 6000 feet, 95 degrees F. The consistent existence of this margin strengthened the T63 manufacturer's decision to show 206 SHP at 6000 feet, 95 degrees F in the 580-E specification.

Prior to the start of the U. S. Army Aviation Test Activity Engineering Tests on the LOH's, referred engine performance curves were obtained from the engine manufacturer. In addition, curves of various correction factors were obtained. These curves were provided to produce 580-E specification power data. A limited check between the 580-E specification and the curves provided showed apparent agreement. As engine calibration data were received from various altitude-temperature combinations, the various parameters were reduced to referred values. These data formed a single curve, as they should, which indicated that the correction factors were reasonable. Later in the test program a more detailed check was made of the 580-E Model Specification. This check revealed that the 580-E specification did not contain power data that agreed with what the actual engines were producing. It also revealed areas of considerable disagreement between the 580-E specification and the manufacturer's referred curves, whereupon another correction curve was furnished. This curve was to be used only when comparing the model specification with actual engine performance. When this new curve was used, there was still disagreement, although not as much as previously.

The manufacturer stated that in order to provide a sufficient power margin to allow the T63-A-5 engine to be put into production and guarantee 206 SHP at 6000 feet on a 95 degree F day, the turbine outlet temperature limit of 1360 degrees F (for takeoff power) would have to be increased to 1380 degrees F.

The engine manufacturer has provided information substantiating the deterioration of power with increasing ambient temperature as presented in this report. This information also further verifies the hot day high altitude power available as presented in this report.

From the discussions with the manufacturer, it was concluded that in order to present a technically accurate and correct picture of the LOH performance, the power deterioration with increasing altitude and temperature as determined from the referred curves would be used.

These curves are presented in Figures No. 63 through 66, Section 3, Appendix I, and are labeled "Engine Model Specification 580-E."

7. <u>Engine Operation</u>

The accumulation of dirt in the compressor section of the T63-A-5 engine installed in OH-6A, USA S/N 62-4212, caused a deterioration in engine performance characteristics. This accumulation was evidenced by an increase in turbine outlet temperature required for a given amount of torque and increased acceleration times from ground idle to flight idle.

Although the engine was operated under atmospheric conditions that were relatively dust free, it was necessary after 75 and 101 engine operating hours to clean the compressor. On both occasions, a relatively large amount of fine silt-like material was removed from the centrifugal portion of the compressor.

Cleaning necessitated removing the engine, splitting the compression case, and brushing the dirt accumulation from inside the centrifugal compressor shroud with a wire brush.

For a clean compressor at approximately 3000 feet density altitude, engine acceleration time was approximately 5-6 seconds. For a dirty compressor under similar conditions, the acceleration time was as long as 15 seconds. At approximately 10,000 feet density altitude, with a dirty compressor, a mild compressor stall was encountered immediately following a rapid power reduction after landing.

[redacted]

APPENDIX III

SYMBOLS AND ABBREVIATIONS

SYMBOL	DEFINITION	UNIT
TAS (Vt)	True Airspeed	Knots
CAS (Vc)(Vcal)	Calibrated Airspeed	Knots
K (Kt)	Knots	Knots
IAS	Indicated Airspeed	Knots
Vne	Never Exceed Airspeed	Knots
Vmax	Maximum Airspeed Attainable	Knots
V_D	Maximum Permissible Dive Speed	Knots
OGE	Out of Ground Effect	-----
IGE	In Ground Effect	-----
C.G.	Center of Gravity	Inches
GW	Gross Weight	Pounds
RPM/rpm	Revolutions per Minute	-----
^oC	Degrees Centigrade	Degrees
^oF	Degrees Fahrenheit	Degrees
SL	Sea Level	-----
SHP	Shaft Horsepower	-----
R/D	Rate of Descent	feet per minute
R/C	Rate of Climb	feet per minute
T/C	Time to Climb	Minutes
C_P	Power Coefficient	-----
C_T	Thrust Coefficient	-----
NAMT	Nautical Air Miles Traveled	-----

NAMPP	Nautical Air Miles Per Pound of Fuel	-----
N_1	Gas Producer Speed	Percent rpm
N_2	Power Turbine Speed	Percent rpm
Hd	Density Altitude	Feet
Hp	Pressure Altitude	Feet
Tt_5 (TOT)	Turbine Outlet Temperature	Degrees
ρ (rho)	Air Mass Density	$\frac{lb\text{-}sec^2}{ft^4}$