UNCLASSIFIED

AD NUMBER

AD909270

NEW LIMITATION CHANGE

TO

Approved for public release, distribution unlimited

FROM

Distribution authorized to U.S. Gov't. agencies only; Test and Evaluation; 24 APR 1973. Other requests shall be referred to Army Test and Evaluation Command, Aberdeen Proving Ground, MD 21005.

AUTHORITY

USAAVSCOM ltr, 12 Nov 1973

THIS PAGE IS UNCLASSIFIED

U.S. ARMY

TEST & EVALUATION COMMAND

USATECOM PROJECT NO. 4-3-0250-31/32/33

PART TWO OF TWO PARTS

REPORT
OF THE ENGINEERING FLIGHT TEST
PERFORMANCE PHASE
OF THE
OH-5A HELICOPTER, UNARMED (CLEAN) and ARMED
WITH THE XM-7 or XM-8 WEAPON SUBSYSTEM

AUGUST 1964

Distribution limited to U.S. Gov't. agencies only; Test and Evaluation; 24 APR 1973 Other requests for this document must be referred to Army Test & Evaluation Comd. attn: AMSTE-SG-H, APG, Md. 21005

HEADQUARTERS

U.S. ARMY AVIATION TEST ACTIVITY

EDWARDS AFB, CALIFORNIA

FOR OFFICIAL USE ONLY

11/25

Copy available to DDC does not permit fully legible reproduction

U. S. ARMY AVIATION TEST ACTIVITY

PART TWO OF TWO PARTS

(6) REPORT

OF THE ENGINEERING FLIGHT TEST

--PERFORMANCE PHASE--

OH-5A HELICOPTER, UNARMED (CLEAN) AND ARMED WITH THE XM-7 OR XM-8 WEAPON SUBSYSTEM, Part II.

(16) USATECOM ~~PROJECT NO.~~ 4-3-0250-31/~~32/35~~

DA ~~PROJECT NO.~~ 1R141803D168

2

LAUREL G. SCHROERS
Project Engineer

EMERY E. NELSON
CWO, U. S. Army
Project Pilot

(11) Aug 64

(12) 109 p.

AUTHENTICATED BY:

Richard J. Kennedy Jr.

RICHARD J. KENNEDY, JR.
Lieutenant Colonel, TC
Commanding

FOR OFFICIAL USE ONLY

D 36 470

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.

TABLE OF CONTENTS

ABSTRACT

Performance flight tests were conducted on the OH-5A helicopter to determine the compliance with the Light Observation Helicopter (LOH) Military Characteristics, to verify the contractual guarantees outlined in the OH-5A Helicopter Model Specification and obtain additional data useful to the Selection Board.

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 auxiliary test sites at Bakersfield and Bishop, California. A total of 99 flights were conducted for 74:25 productive flight hours. These tests were accomplished during the period of 20 February 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-5A helicopter met all contractual performance guarantees except for the out-of-ground effect (OGE) hovering performance at 6000 feet, normal gross weight on a 35 degree Centigrade (C) day. The hovering ceiling under these conditions was 3300 feet.

The OH-5A helicopter met the Military Characteristics requirement for 3 hours endurance with a 400 pound payload.

The OH-5A did not meet the Military Characteristics requirements for cruise speed, endurance at 85 percent cruise power, OGE hot day hover performance at 6000 feet or OGE hover performance at overload gross weight, sea level standard day.

The OH-5A helicopter takeoff, climb and autorotational performance was satisfactory for a helicopter of this power and weight class.

The normal gross weight limit airspeed (Vne) established for the OH-5A helicopter was satisfactory up to an altitude of approximately 9500 feet. At approximately 10,000 feet, operation at a minimum power-on rotor speed of 353 rpm (96 percent) produced a vibration limit airspeed that was less than Vne. At approximately 14,700 feet, a vibration limit airspeed less than Vne was encountered even though the helicopter was operated at maximum power-on rotor speed of 368 rpm (100 percent). The overload gross weight Vne was satisfactory under all conditions tested.

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.

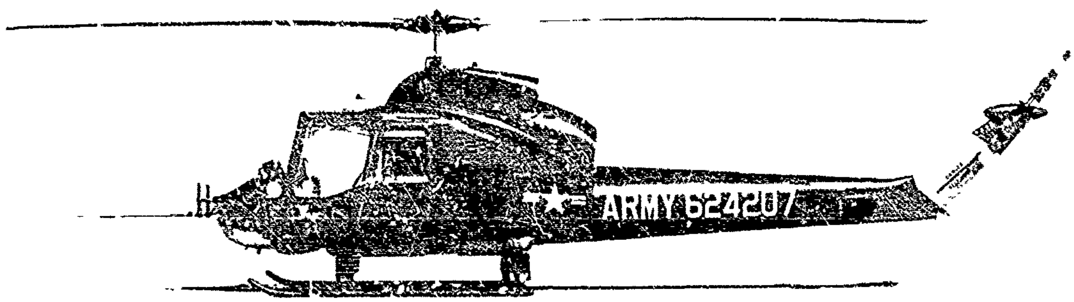


PHOTO 1 – OH-5A

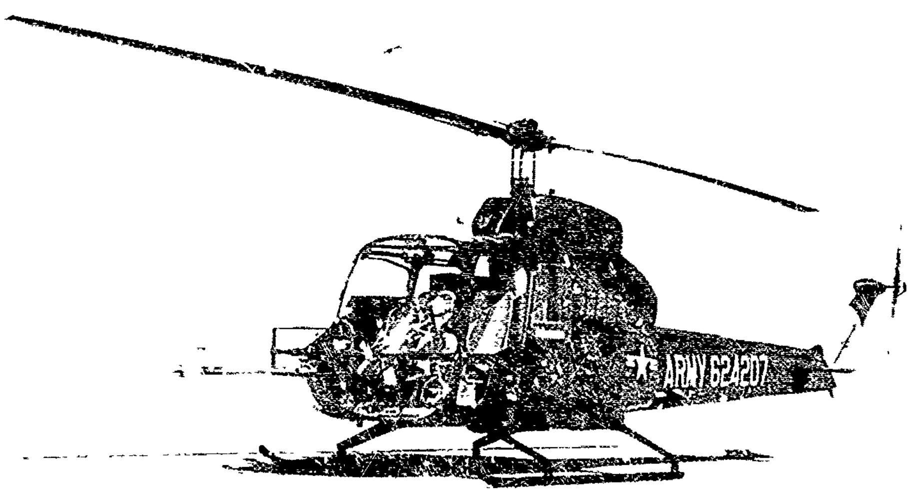


PHOTO 2 – OH-5A

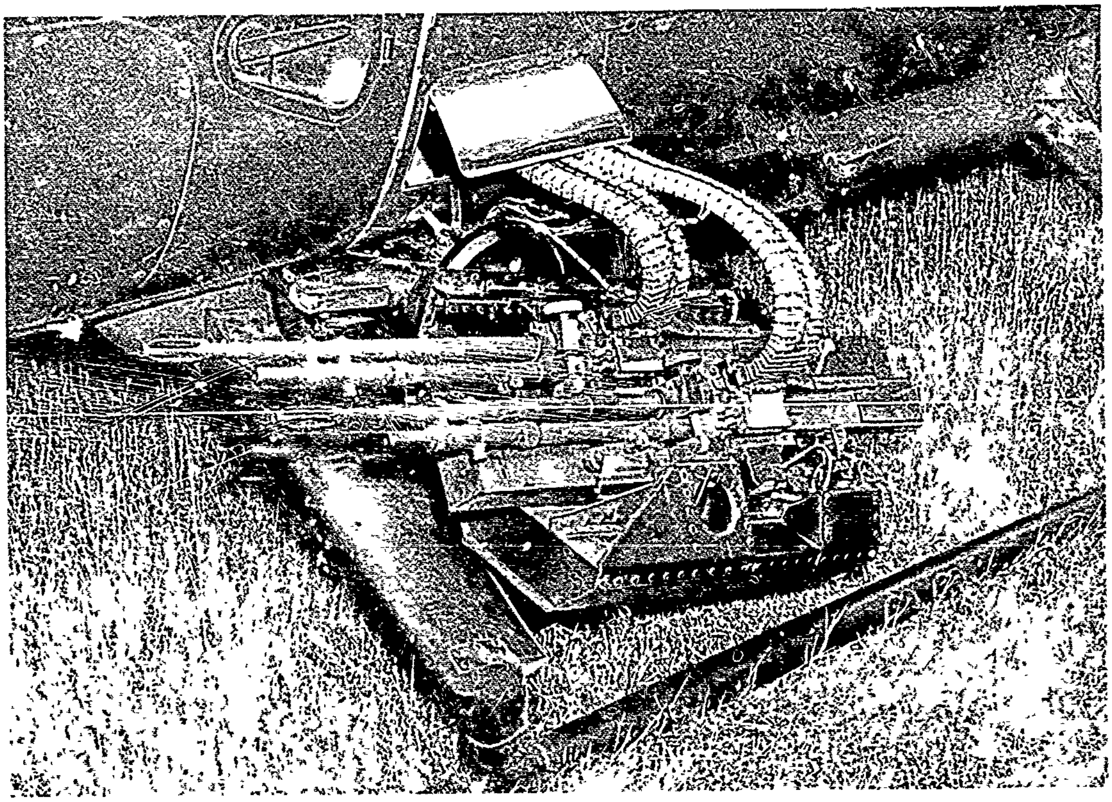


PHOTO 3 – XM-7 ARMAMENT KIT

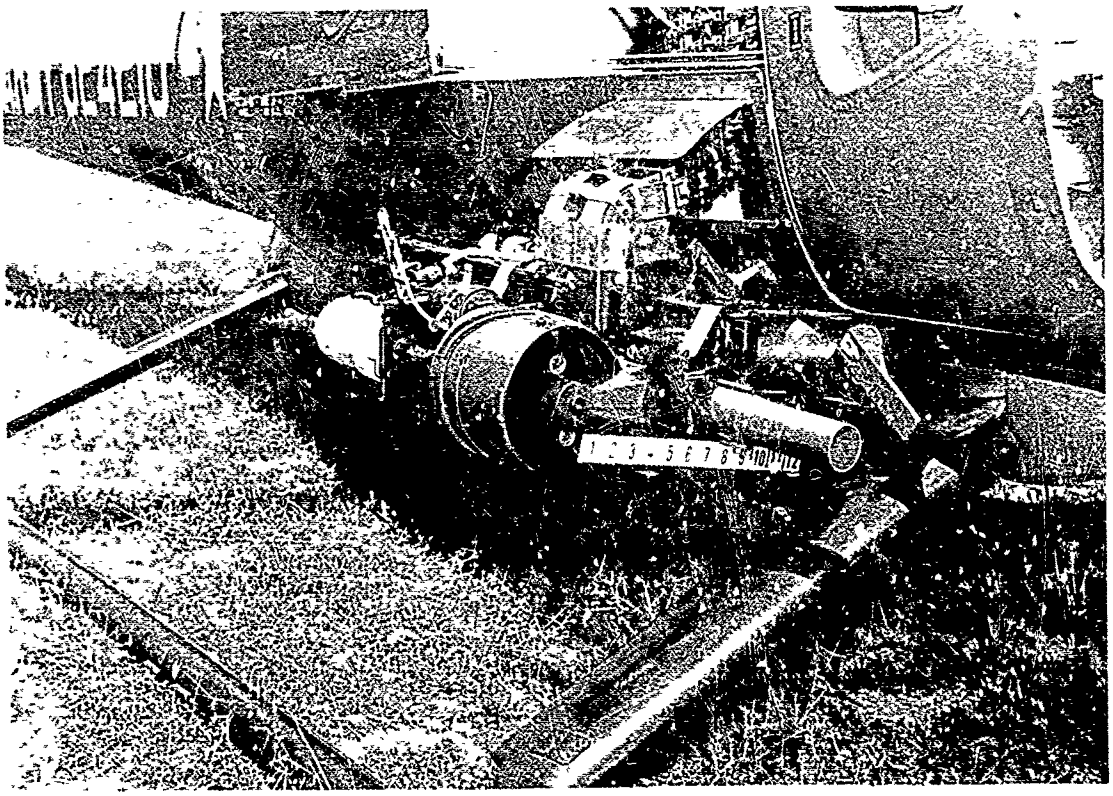


PHOTO 4 – XM-8 ARMAMENT KIT

SECTION 1 - GENERAL

1.1 REFERENCES

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

m. Preliminary Report, "Flight Evaluation of the T63-A-5 Gas Turbine Engine Installed in the UH-13R Helicopter," USATECOM Project No. 4-4-0240-02, U. S. Army Aviation Test Activity, January 1964.

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

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

p. Final Report of "OH-13H Gross Weight Increase/XM-1 Armament Kit Performance Test," U. S. Army Aviation Test Activity, June 1963.

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-5A to (a) confirm contractor compliance with the approved Army Military Characteristics for an unarmed (clean) and armed OH-5A 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; Part I, Section 3, Appendix II; and Part II, Section 3, Appendix II.

1.6 BACKGROUND

See Part I, Section 1.

1.7 FINDINGS

The OH-5A Model Specification performance guarantees and the requirements stated in the Light Observation Helicopter Military Characteristics were evaluated using power based on Engine Model Specification No. 580-A.

The OH-5A helicopter performance complies with all contractual guarantees except for out-of-ground effect (OGE) hover performance at 6000 feet with a normal gross weight of 2530 pounds on a 35 degree Centigrade (C) day. The hover ceiling under these conditions was 3300 feet.

The OH-5A helicopter performance satisfied only one of the five requirements of the Military Characteristics and this was the requirement for a 400 pound payload with 3 hours of fuel. Comparison of the OH-5A helicopter performance with the requirements stated in the Military Characteristics revealed the following requirements were not met: The requirement for a 110 knot cruise speed at normal gross weight, sea level standard day was not met by 1 knot. The requirement for an endurance of 3 hours at 85 percent cruise power, normal gross weight, sea level standard day was not met by 6 minutes. The requirement for an OGE hovering capability at an overload gross weight of 3000 pounds at sea level on a standard day was not met by 40 pounds. The requirement for an OGE hover capability at 6000 feet at normal gross weight on a 35 degree C day was not met by 2700 feet.

The limit airspeed (Vne) established for the OH-5A helicopter was satisfactory up to an altitude of approximately 9500 feet. At approximately 10,000 feet, operation at a minimum power-on rotor speed of 353 rpm (96 percent) produced a vibration limited airspeed less than Vne. At approximately 14,700 feet, a vibration limited airspeed below Vne was encountered, even though the maximum power-on rotor speed of 368 rpm (100 percent) was used. The overload gross weight (any weight over 2530 pounds) Vne was satisfactory under all conditions tested.

The takeoff, climb and autorotational performance of the OH-5A was satisfactory for a helicopter of this power and weight class.

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-5A 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 99 flights were conducted for 74:25 productive flight hours. The tests were accomplished during the period of 26 February 1964 to 1 July 1964.

Level flight performance data were obtained at Edwards Air Force Base and at the sea level site at Bakersfield, California. Climb performance data were not obtained entirely at Bakersfield where the climbs could be initiated at 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 meaningful performance data could be obtained. The test data were recorded by hand from sensitive instruments or automatically using a photo panel.

Power available and fuel flow as specified in the T63-A Engine Model Specification 580-A were used to check the contractual guarantee and the desired performance defined in the Military Characteristics for hovering, maximum airspeed and endurance.

Summary performance was obtained using power available and fuel flow information based on the Engine Model Specification No. 580-E on a sea level standard day. Engine performance at the 6000 foot altitude, 95 degree Fahrenheit (F) ambient temperature condition and at all other altitude-temperature combinations was 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.

Using the power available as described above, it was determined that a T63-A-5 engine, which will just meet the sea level standard day power, will not produce the 206 shaft horsepower (SHP) at 6000 feet, 95 degrees F as shown in the 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).

2.1 TAKEOFFS

2.1.1 OBJECTIVE

Takeoff tests were conducted to determine the performance of the OH-5A 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 accomplished at two altitudes and several gross weights to define the performance over as wide an envelope as possible. Takeoff performance was evaluated using two takeoff techniques which are commonly used in the field. 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 used for each series of takeoffs varied from the minimum achievable to maximum practical airspeed (approximately 50 knots indicated airspeed (KIAS)). All takeoffs were performed in winds of 3 knots or less. These tests were conducted at density altitudes of approximately 5000 and 10,000 feet. Gross weight was varied from 2330 to 3000 pounds at a mid center-of-gravity (C.G.) location (Station 98.5). All takeoff tests were conducted with a main rotor speed of 368 rpm (100 percent power turbine speed (N_2)) using takeoff power.

2.1.3 RESULTS

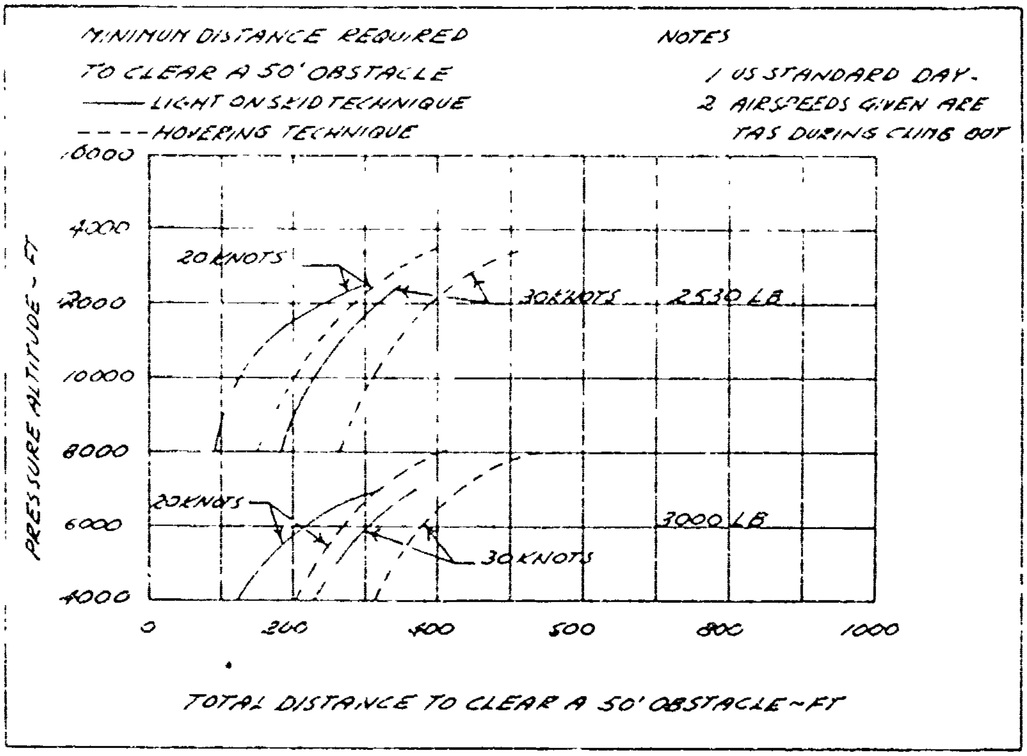
Takeoff test results are graphically presented in Figures No. 1 through 18, Section 3, Appendix I.

2.1.4 ANALYSIS

a. Comparative Techniques

Two techniques were used during this evaluation to obtain quantitative data. The first technique consisted of a level acceleration from a stabilized 2 foot hover. The helicopter was accelerated at the hover skid height to an airspeed slightly below the aim climb-out airspeed, rotated to climb attitude and climb out was accomplished at the desired airspeed. The second technique consisted of a simultaneous climb and acceleration initiated from a "light on the skids" condition. Sufficient power was applied to maintain the helicopter light on the skids and takeoff was initiated by applying takeoff power available. As lift-off occurred, a pitch attitude was selected and held constant to obtain the desired airspeed at 50 feet. Once airborne, altitude and airspeed were increased simultaneously.

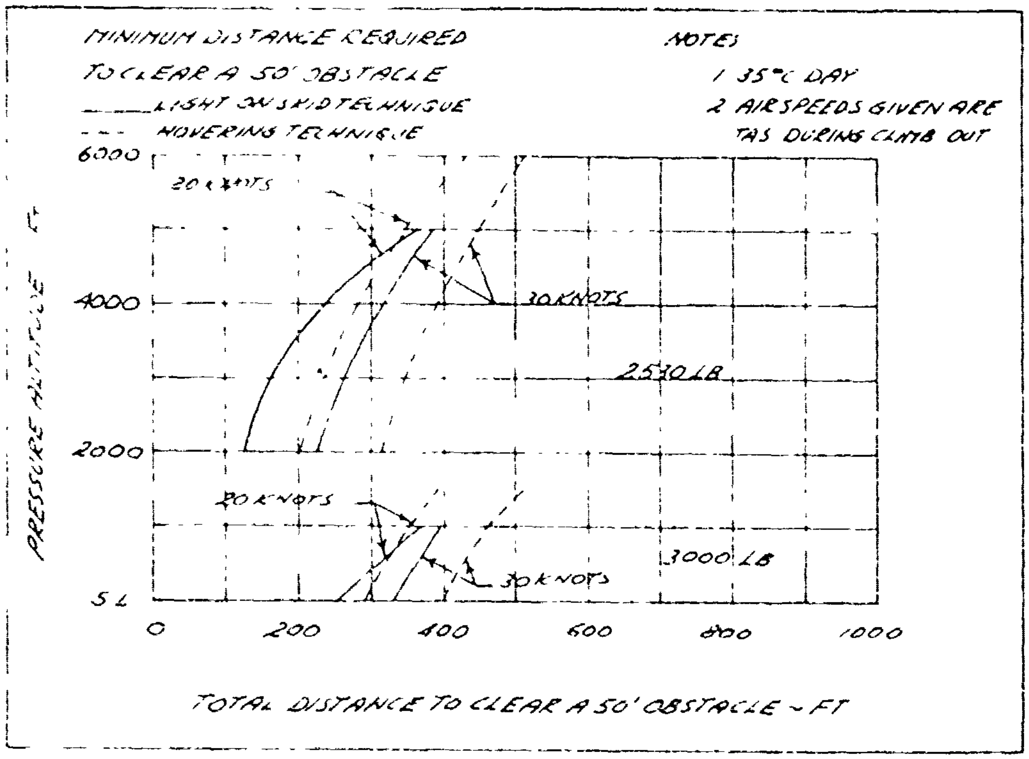
A comparison of the two techniques on a standard day and on a 35 degree Centigrade (C) day is presented in the following two figures:



(See Page 6 for second figure).

The data presented in these two figures were obtained from cross plot of Figures No. 1, 2, 10 and 11, Section 3, Appendix I, which were calculated from test data and installed power available based on Engine Model Specification 580-E.

The light on the skids technique was optimum when takeoff conditions (altitude, gross weight, ambient temperature) enabled the helicopter to hover at a skid height of 3 feet or more. The hover



technique became optimum under conditions of low excess power. Under these conditions, the level acceleration technique provided maximum advantage of the available ground effect to assist acceleration to translational speeds.

b. Flight Characteristics During Takeoff

Maximum performance takeoffs were achieved by anticipating the helicopter's flight characteristics and coordinating control movements to control these characteristics. When using the hovering technique, the helicopter accelerated in a "nose-low" attitude and had a tendency to settle (especially when only a small amount of excess power was available) just prior to the translational lift region. Careful control manipulation had to be exercised to level the helicopter sufficiently to maintain the desired 2 foot skid height without affecting appreciably the established acceleration. As the helicopter passed through the translational lift region, the nose had a tendency to pitch up and forward cyclic control was required to maintain the desired altitude that would most quickly accelerate the helicopter to the desired climb-out airspeed. Rotation from the accelerating attitude to the climb attitude had to be a coordinated control input so that the desired climb-out airspeed could be realized. To accomplish this, the rotation was initiated approximately 4 to 8 knots below the desired climb-out airspeed. The higher climb speeds required greater lead. Good pilot judgement was required to accomplish this with a minimum of under or overshoot. The pilot had to estimate the acceleration rate, and from experience, determine when the rotation should be initiated. The control movements that were required during the acceleration are normal for a helicopter of this type.

An undesirable airframe pitching oscillation was encountered when performing level acceleration takeoffs when a small amount of excess power was available. This airframe oscillation persisted for 3 to 8 cycles depending on climb-out airspeed (increasing with increasing speed) and was caused by the transmission rocking longitudinally in the gimble mounts and transmitting the motion to the airframe through the bottom snubber block.

When performing takeoffs using the light on the skids technique, the pitch attitude was the basic reference of forward speed. Good pilot judgement was necessary to increase airspeed and altitude simultaneously. When power available was insufficient to hover above 2 to 3 feet, there was an appreciable decrease in rate of climb above 10 to 15 feet during climb-out until sufficient forward speed was obtained. In some cases, the helicopter would level off or even descend prior to obtaining translational lift.

A third technique was used to obtain qualitative data whenever power available was insufficient to hover. Takeoffs were easily accomplished by applying forward cyclic control and sliding the helicopter on the skid gear until lift-off occurred - somewhere between 5 and 10 knots. The pitch attitude was near level throughout forward movement and lift-off. The helicopter lifted off smoothly and immediately began to climb and accelerate, requiring forward cyclic control to accelerate in ground effect (IGE) to a safe climb-out airspeed.

During all takeoffs, oscillations in the aircraft standard airspeed system made it unreliable in the 15 to 20 knot range and unusable below 15 knots. At approximately 10 knots forward speed, the system read negative, for which it cannot be calibrated.

Under conditions in which there was a large amount of excess power available, the acceleration through the translational lift region was accomplished with little or no vibration. The only definite indication that the helicopter had passed through the translational lift region was a decrease in left pedal required. As the excess power available to hover decreased, the translational region became more pronounced. This was characterized by difficulty in maintaining the desired skid height (level acceleration technique) or the desired rate of climb (climb and accelerate technique). On passing through the translational lift region, there was a slight increase in vibration level, a nose-up pitching tendency and a definite decrease in left pedal requirement.

c. Power Management

A rapid application of excess power was desirable when evaluating maximum takeoff performance. Power management was difficult when there was a large amount of excess power available because of the T63-A-5 engine acceleration characteristics with respect to torque buildup. The rate of torque buildup varied from a low rate of torque increase at the initiation of the power application to a high rate of torque increase at the termination of the power application. This change in rate of torque increase occurred within the 1 to 2 seconds required to apply collective pitch and, consequently, made it difficult to coordinate collective, longitudinal and pedal control movements.

The engine power turbine speed (N_2) and main rotor speed are controlled by the N_2 governor. An N_2 selector "beep" switch was located on the collective pitch control. This switch was used to energize an actuator on the N_2 governor input lever. As stated by the airframe manufacturer, the "beep" actuator during a ground check, should be capable of varying rotor speed by 6 percent when the collective pitch control is positioned to obtain 36 percent engine output shaft torque. This was referred to as the "beep" range and the minimum and maximum limits were 95.5 percent and 101.5 percent respectively. On the test helicopter, this range shifted numerous times and required ground reajustments to insure adequate "beep" range to maintain the desired rotor speed. This became critical when operating at the overload gross weight condition (3000 pounds) at which the rotor speed is restricted to 368 rpm (100 percent N_2). The cause of this shift in "beep" range was not determined during this evaluation and the only corrective action taken was to adjust the N_2 governor to obtain the desired "beep" range.

2.2 HOVER

2.2.1 OBJECTIVE

Hovering tests were conducted to determine hovering performance of the OH-5A 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 utilizing the free flight method with data being recorded at various pre-selected skid heights both in-ground (IGE) and out-of-ground effect (OGE). A weighted cord of the desired length was used with hand signals and radio instructions from a ground observer to obtain precisely the desired skid height. 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 98.5) and rotor speeds of 353, 360 and 368 rpm. Testing was accomplished at density altitudes from sea level to 11,100 feet.

Hovering tests without ground reference were conducted at density altitudes of 5600 to 10,600 feet. The hovering condition was established by reference to a weighted line, approximately 150 feet in length, suspended below the helicopter. Zero airspeed was obtained when no bow was present in the line. A constant altitude was maintained by reference to a sensitive rate of climb indicator and an altimeter. These tests were conducted at a gross weight of 2470 pounds, at a mid C.G. location (Station 98.5) and a rotor speed of 368 rpm.

2.2.3 RESULTS

Hovering test results are presented graphically in Figures No. 19 through 28, Section 3, Appendix I.

2.2.4 ANALYSIS

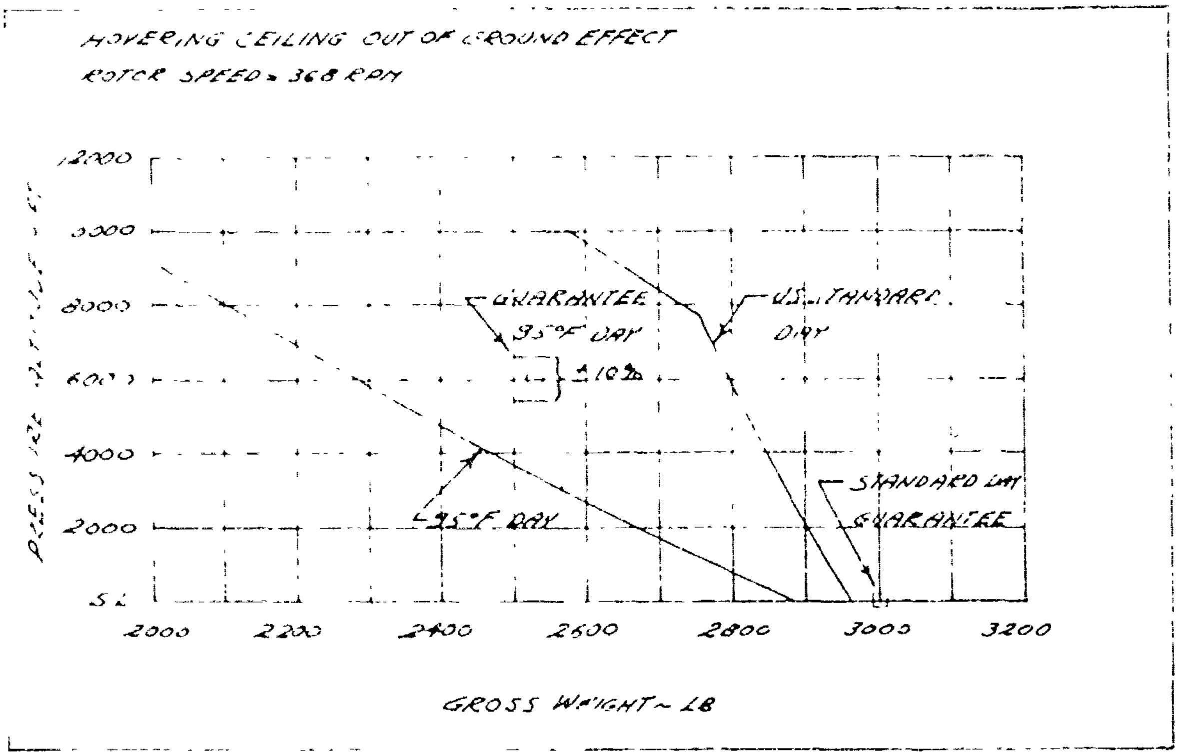
a. Hovering Guarantee

The OH-5A helicopter will not meet the OGE hovering performance guarantee of 6000 feet on a 95 degree Fahrenheit (F) day at normal gross weight (2530 pounds). Using installed takeoff power as obtained from Engine Model Specification 580-A at 95 degrees F, the OH-5A will hover at 3300 feet at the normal gross weight. This performance is approximately 45 percent short of the guarantee.

The hovering performance based on Engine Model Specification No. 580-A for a standard day and a 95 degree F day is presented in the figure on Page 10.

On a sea level standard day, using takeoff power based on Engine Model Specification No. 580-A (torque limited at 250 shaft horsepower (SHP)), the OH-5A helicopter will hover OGE at 2960 pounds. This weight is 98.7 percent of the overload gross weight (3000 pounds); therefore, the OH-5A helicopter met this portion of the hovering guarantee (Guarantee states 3000 pounds $\pm$ 5 percent).

The hovering performance required in the Military Characteristics is the same as the guarantee for the OH-5A helicopter with the exception of the $\pm$ 5 percent margin.



b. Hovering Performance

The summary hovering performance presented in Figure No. 19, Section 3, Appendix I, is based on installed takeoff power as obtained from Engine Model Specification No. 580-E (torque limited at 275 SHP). Using this power, the OH-5A helicopter could hover OGE on a 35 degree Centigrade (C) day at the normal gross weight (2530 pounds) at an altitude of 3000 feet. This performance is approximately 50 percent below the guarantee. On a sea level standard day, the OH-5A helicopter could hover OGE at 3160 pounds. This weight is 105 percent of the overload gross weight and, therefore, exceeds the guarantee.

The IGE (2 foot skid height) hover ceiling was determined from test data and standard day takeoff power available based on Engine Model Specification No. 580-E. The OH-5A helicopter had an IGE hover ceiling of 13,700 feet at normal gross weight (2530 pounds) and 8100 feet at the overload gross weight (3000 pounds).

The difference in torque limits (25 SHP) between Engine Model Specifications No. 580-A and No. 580-E is the reason that hovering on a sea level standard day is significantly improved when based on the 580-E specification. On a 35 degree (C) day and at 6000 feet, the T63-A-5 engine was not torque limited but temperature limited. Since a 580-A specification engine has more power than a 580-E specification engine at limit turbine outlet temperature, the helicopter hovering performance was better.

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 5 minutes, were also conducted from sea level at one gross weight. 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 at 2530 pounds (normal) and 3000 pounds (overload) at a mid center-of-gravity (C.G.) location (Station 98.5) and a rotor speed of 368 rpm.

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

2.3.3 RESULTS

Climb results are presented graphically in Figures No. 30 through 33, Section 3, Appendix I.

2.3.4 ANALYSIS

The OH-5A helicopter at the normal gross weight (2530 pounds) with takeoff power on a standard day had an initial rate of climb at sea level of 1850 feet per minute. At the engine critical altitude of 6200 feet, the rate of climb on a standard day was 1720 feet per minute. At the end of the 5 minute limit for takeoff power, the altitude was 8700 feet and the rate of climb was 15[illegible]0 feet per minute.

Climbs conducted from sea level to service ceiling (altitude at which the rate of climb equals 100 feet per minute) at the normal gross weight, using maximum continuous power, produced an initial rate of climb of 1350 feet per minute and at the engine critical altitude (7800 feet) a rate of climb of 1300 feet per minute. Under these conditions, the OH-5A helicopter had a service ceiling of 17,200 feet.

Climbs at the overload gross weight (3000 pounds) using maximum continuous power on a standard day produced an initial rate of climb at sea level of 1050 feet per minute. The rate of climb for the same conditions at the engine critical altitude (7800 feet) was 870 feet per minute.

Additional climbs were conducted at the normal and overload gross weight using a climb speed schedule that was slightly higher than that developed from the level flight data and sawtooth climbs. This higher climb speed schedule produced a decrease in climb performance (See Figures No. 32 and 33, Section 3, Appendix I).

The following table presents a comparison of the climb performance of the OH-5A helicopter using maximum continuous power with an OH-13H helicopter:

TABLE I

Aircraft	Gross Weight lb	Rate of Climb Sea Level-fpm	Service Ceiling ft
OH-5A	2530	1350	17,200
	3000	1050	--
OH-13H	2550	1080	13,050
	2750	880	11,050

The flight characteristics of the OH-5A helicopter are generally satisfactory. The climb speed schedule was easy to maintain when using maximum continuous power. When using takeoff power, the climb speed schedule was difficult to maintain at the lower altitudes with the helicopter loaded to a light gross weight. Under these conditions, a 1 to 2 knot variation in airspeed was noticed without a detectable change in attitude. When a large amount of excess power is available, as in the condition described above, a very slight change in attitude (perhaps undetectable) could produce this variation in airspeed. Another factor that contributes to this problem was that power management was more difficult when the engine was torque limited but became relatively easy when power was limited by turbine outlet temperature.

Approximately 3000 to 4000 feet below service ceiling during the maximum continuous power climb at normal gross weight, a marked increase in vibration as well as a decrease in overall control was noted. This was attributed to the onset of blade stall. At the service ceiling, the overall feel of the helicopter was uncomfortable and it appeared as if complete control was not available or could not be maintained. Lateral control sensitivity increased with altitude (See Part I, Section 2, paragraph 2.6.4.2) with right roll sensitivity increasing more than left roll sensitivity. This change in sensitivity produced a condition that caused the pilot to over-control and thereby created a feeling that complete control was not available.

2.4 LEVEL FLIGHT

2.4.1 OBJECTIVE

Tests were conducted in level flight to determine the range endurance, speed and power required of the OH-6A helicopter. Tests with the XM-7 and XM-8 weapons systems installed were also conducted to determine their effect on helicopter performance. Data from these tests were used to check 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 (XM-7 or XM-8) 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 flight at various airspeeds throughout the allowable speed range at approximately 10 knot increments so as to adequately define 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 weight was varied from 2160 to 2940 pounds at a mid center-of-gravity (C.G.) location (Station 98.5). Rotor speeds of 368 and 353 rpm were used.

Two additional tests were conducted, one at a forward C.G. location (Station 95.8) and one at an aft C.G. location (Station 100.7). Tests in the armed configurations were conducted at density altitudes of approximately [illegible] and 4300 feet, at a gross weight of 2540 pounds, at a mid C.G. location (Station 98.5) and a rotor speed of 368 rpm.

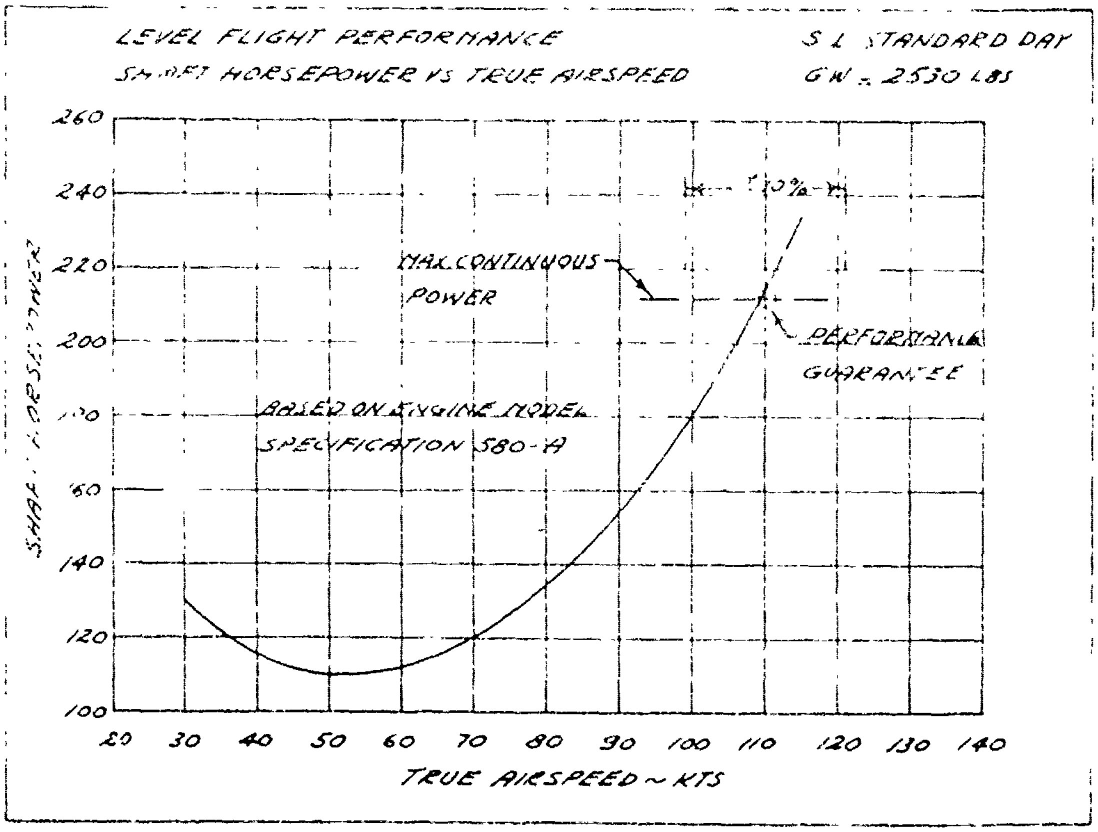
2.4.3 RESULTS

Level flight performance test results are graphically presented in Figures No. 34 through 52, Section 3, Appendix I.

2.4.4 ANALYSIS

a. Guarantees

The OH-5A helicopter meets the level flight sea level standard day maximum airspeed (Vmax) guarantee. The model specification requires a maximum airspeed of 110 knots ± 10 percent at the normal gross weight and normal rated power. The curve presented in the following figure is based on sea level standard day conditions, a design gross weight of 2530 pounds and installed maximum continuous power available obtained from Engine Model Specification No. 580-A.



FOR OFFICIAL USE ONLY

As shown in the figure on the preceding page, the OH-5A helicopter had a Vmax of 109 knots which is within the ± 10 percent margin of 110 knots. The Military Characteristics specify a cruise speed of 110 knots.

The OH-5A helicopter meets the sea level standard day endurance guarantee. The model specification requires an endurance of 3 hours, ± 10 percent, based on sea level standard day and the conditions listed below:

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

(2) Cruise: 102 knots at sea level.

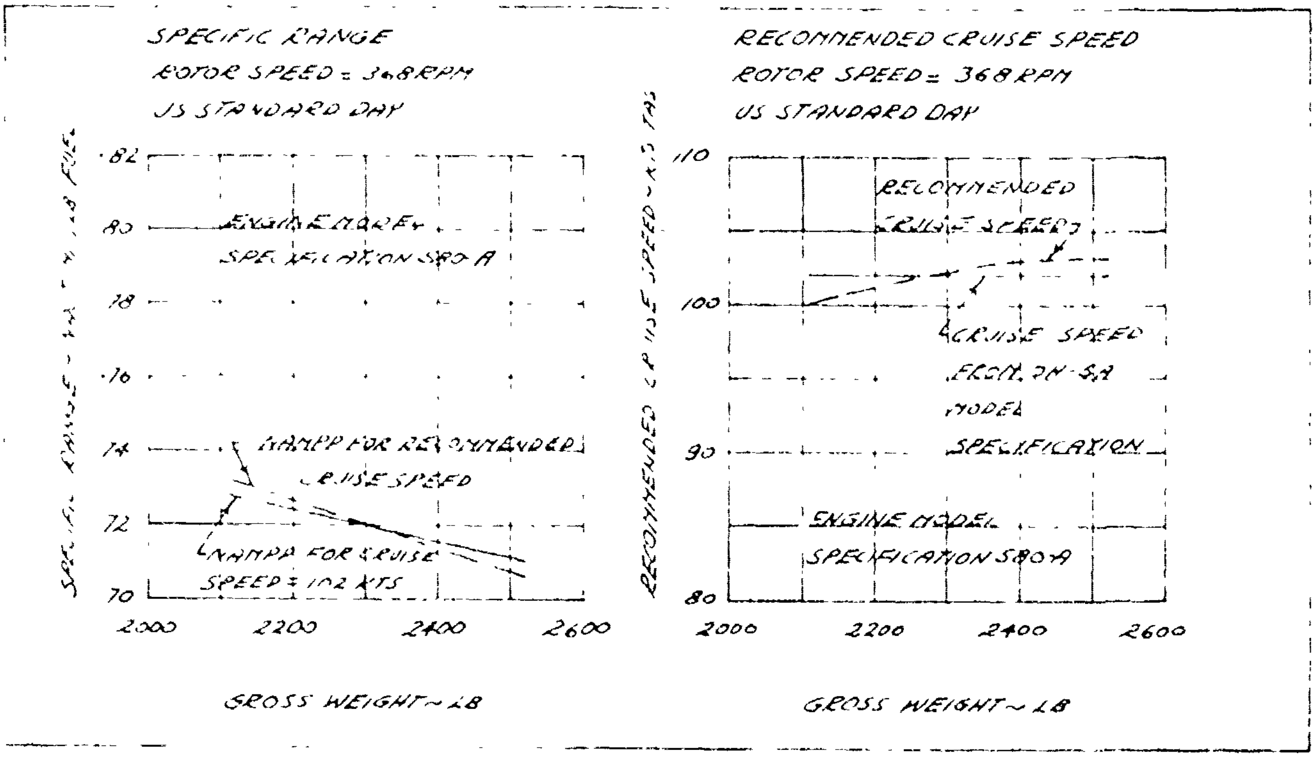
(3) Reserve: None.

(4) Useful load at normal gross weight:

Pilot (1)	200 pounds
Observer and Cargo	400 pounds
Fuel (JP-4) (as necessary for endurance problem)	
Oil (as necessary for above fuel)	

The Federal Aviation Agency Type Certificate (FAA TC) states that 16.25 pounds of fuel are unusable in some flight attitudes. This amount of fuel, therefore, was subtracted from total fuel on board and added to the empty weight when calculating the endurance problem. Under the conditions described above, the OH-5A helicopter had an endurance of 2.85 hours, which is within the ± 10 percent margin of 3 hours. The endurance would not be significantly increased by varying airspeed as gross weight decreased because the recommended cruise speed presented in the following figure produced a curve of specific range versus gross weight that is essentially the same as that obtained from the 102 knot cruise speed required by the model specification. (See Page 16 for figure).

The Military Characteristics required an endurance of 3 hours at 85 percent of power required to cruise at 110 knots. In the case of the OH-5A helicopter at normal gross weight a cruise speed of 110 knots is not attainable on a sea level standard day when using maximum continuous power based on Engine Model Specification No. 580-A. It will, however, cruise at 110 knots at a lighter gross weight. The Military Characteristics requirement, therefore, was verified by calculating endurance based on 85 percent of 212 shaft horsepower (SHP) or 85 percent of the power required to cruise at 110 knots, whichever was less. Under these conditions, the OH-5A helicopter had an endurance of 2.90 hours.



The Military Characteristics also required an endurance that is based on sea level standard day, a payload of 400 pounds and fuel for 3 hours (cruise speed or cruise power not specified). The cruise speed for this endurance problem was selected to correspond to minimum power required for level flight. The OH-5A helicopter had an endurance under these conditions of 3.92 hours based on Engine Model Specification No. 580-A.

b. Performance

In general, the level flight handling characteristics are satisfactory. The limit airspeed (Vne) as established by the FAA TIA for normal gross weight (2530 pounds) was adequate up to approximately 9500 feet for all permissible power-on rotor speeds. A vibration limiting airspeed to a value less than Vne was first encountered when operating at 10,000 feet and minimum power-on rotor speed of 353 rpm (96 percent). (See Figure No. 43, Section 3, Appendix I). At approximately 14,700 feet the established Vne could not be obtained because of vibrations even through the helicopter was operated at the maximum power-on rotor speed of 368 rpm (100 percent), (See Figure No. 44, Section 3, Appendix I.) When operating at the overload gross weight (any weight above 2530 pounds) the rotor speed is restricted to 368 rpm (100 percent) and the established Vne was satisfactory at the conditions tested.

The antenna mounted on the nose of the aircraft has negligible effect on level flight performance as shown in Figures No. 45 and 46, Section 3, Appendix I.

Level flight performance was affected by C.G. location. One flight was flown at a density altitude of 5000 feet with the helicopter loaded to the normal gross weight with a forward C.G. location (Station 95.8). At the recommended cruise speed of 109 knots true airspeed (KTAS) (.99 Max nautical air miles per pound of fuel (NAMPP)) this forward C.G. location produced an increase in equivalent flat plate area of approximately 1.13 square feet. The specific range will decrease when cruising at a forward C.G. location. The flight characteristics under these conditions, however, are improved (less vibration) and offset the slight improvement in range that is realized by cruising at a mid C.G. location. Level flight performance at an aft C.G. location (Station 100.7) was essentially the same as at a mid C.G. (Station 98.5).

Level flight performance in the armed configuration was compared to level flight performance in the clean configuration for the same test conditions. At the recommended cruise speed (.99 max specific range) the XM-7 machine gun produced an equivalent flat plate area of approximately 2.4 square feet at a density altitude of 1230 feet and approximately 2.9 square feet at a density altitude of 4530 feet. The XM-8, at the recommended cruise speed, produced an equivalent flat plate area of approximately 1.8 square feet at a density altitude of 1620 feet and approximately 2.0 square feet at 4270 feet. The equivalent flat plate area for both armament kits decreased as airspeed increased because the kits were mounted on the aircraft so that in the stowed position, there was a minimum effect on level flight performance at cruise speed. The installation of the XM-7 armament kit caused a reduction in recommended cruise speed of 6.5 KTAS. The reduction in cruise speed with XM-8 installed was 3.5 KTAS.

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 airspeed and rotor speed combinations throughout their allowable ranges. During the descent, time and altitude were recorded so that rate of descent could be determined.

A series of stabilized descents was conducted at a rotor speed of 368 rpm and at various airspeeds between 35 and 85 knots calibrated airspeed (KCAS). From these tests the airspeed for minimum rate of descent was determined. At that airspeed, a second series of descents was conducted at rotor speeds from 345 to 390 rpm.

The autorotational descents were conducted at altitudes from approximately 10,000 to 5000 feet. Various gross weights from 2275 to 2915 pounds were flown at a mid center-of-gravity C.G. location.

2.5.3 RESULTS

The results of autorotational descents are presented graphically in Figures No. 53 through 55, Section 3, Appendix I.

2.5.4 ANALYSIS

The OH-5A helicopter had a minimum rate of descent of 1420 feet per minute at 55 KCAS and a rotor speed of 355 rpm. At a density altitude of 5000 feet, the true airspeed for maximum angle of descent was determined to be 79 knots at a rotor speed of 368 rpm. Rate of descent was not affected by altitude or gross weight. It was not possible to obtain a true autorotational condition below approximately 340 rotor rpm with the engine operating at ground idle. Variation in rotor speed between 340 and 360 rpm (92.4 to 98.7 percent) did not appreciably affect the rate of descent.

The OH-5A helicopter flight characteristics during autorotation and the engine response during autorotational recoveries were determined to be essentially the same as stated in Part I, Section 2, paragraphs 2.8.4 and 2.8.5.

2.6 <u>AIRSPEED CALIBRATION</u>

2.6.1 OBJECTIVE

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

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 25 to 115 knots calibrated airspeed (KCAS) using approximately 10 knot increments were flown. These tests were conducted at a density altitude of 460 feet, a gross weight of 2500 pounds, at a mid center-of-gravity (C.G.) location (Station 98.6) and a rotor speed of 368 rpm.

2.6.3 RESULTS

Results of the airspeed calibration are graphically presented in Figures No. 70 and 71, Section 3, Appendix I.

2.6.4 ANALYSIS

The position error of the standard system was acceptable for climb and level flight above airspeeds of 20 KCAS. The largest position error occurred at Vne where a zero position error is desirable. (See Figure 71, Section 3, Appendix I). The erratic operation of the standard system at airspeeds below 20 knots and its effect on takeoff performance is discussed in Section 2.1.4.

SECTION 3 - APPENDICES

APPENDIX I - TEST DATA

FIGURE NO. 1

SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER
GROSS WEIGHT = 2530 LB.
ROTOR SPEED = 368 RPM
CLEAN CONFIGURATION
TAKE-OFF POWER.

NOTE.
DERIVED FROM FIGURES
3, 26 AND 56

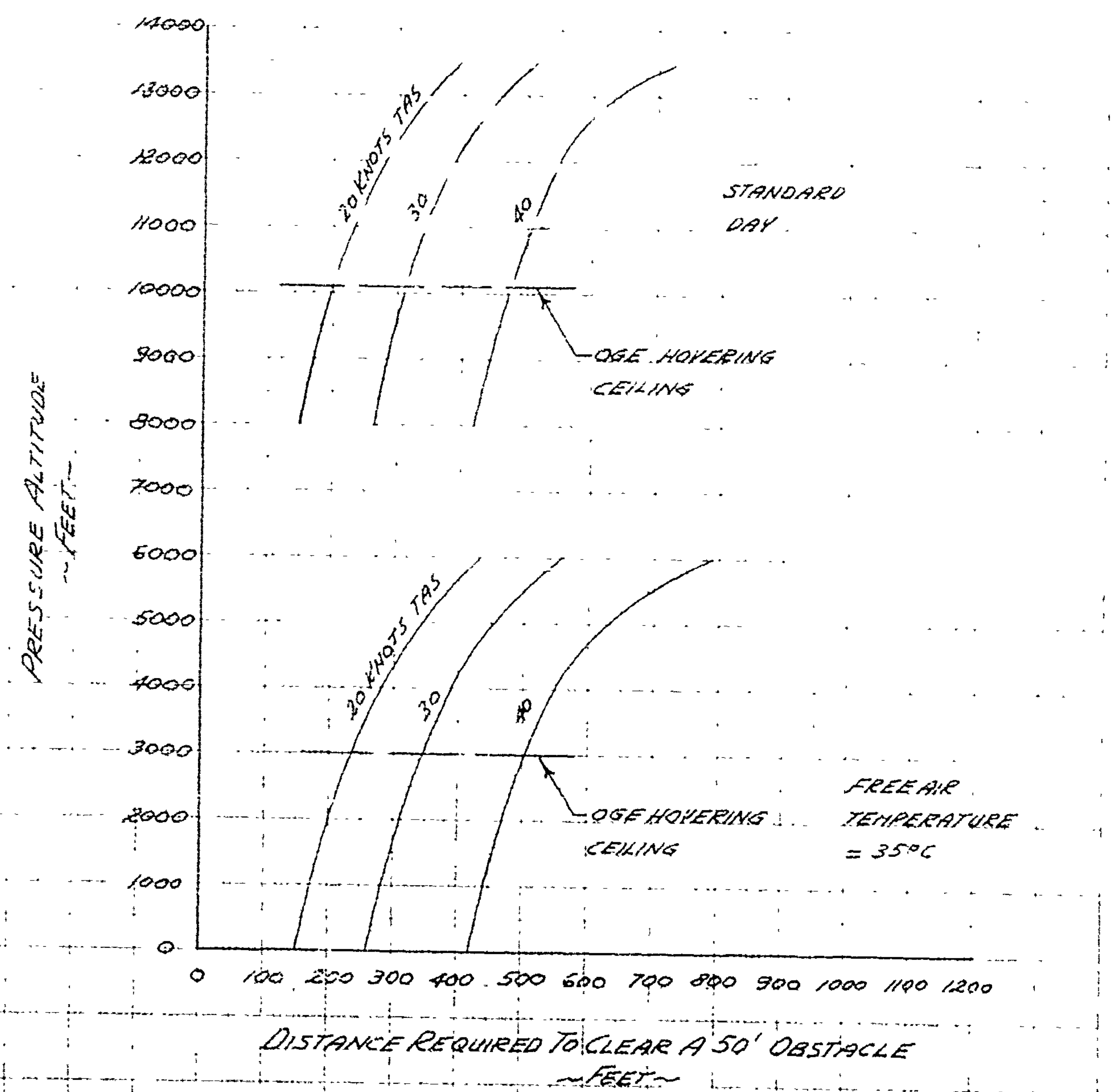
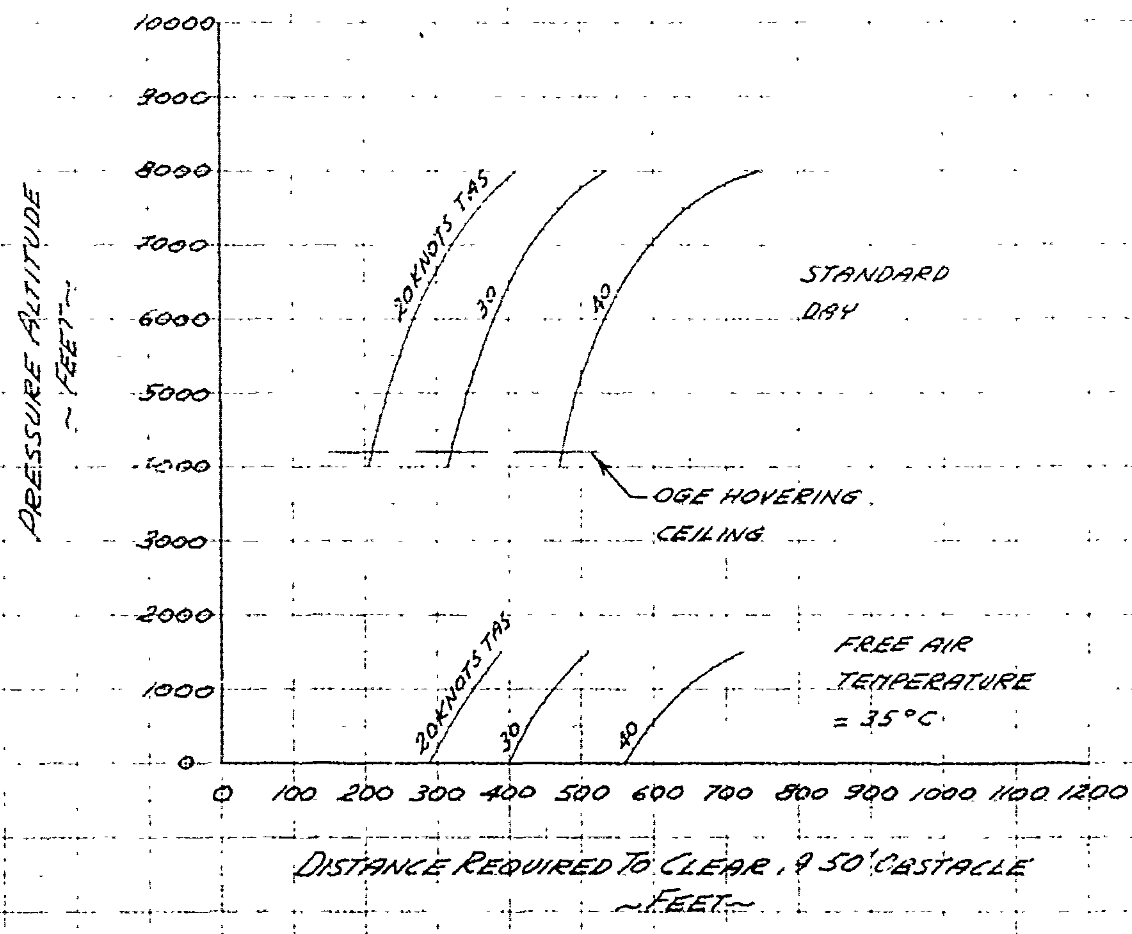


FIGURE No. 2
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER
GROSS WEIGHT = 3000 LB.
ROTOR SPEED = 368 RPM
CLEAN CONFIGURATION
TAKE-OFF POWER.

NOTE.
DERIVED FROM FIGURES
3, 26 AND 56



FOR OFFICIAL USE ONLY

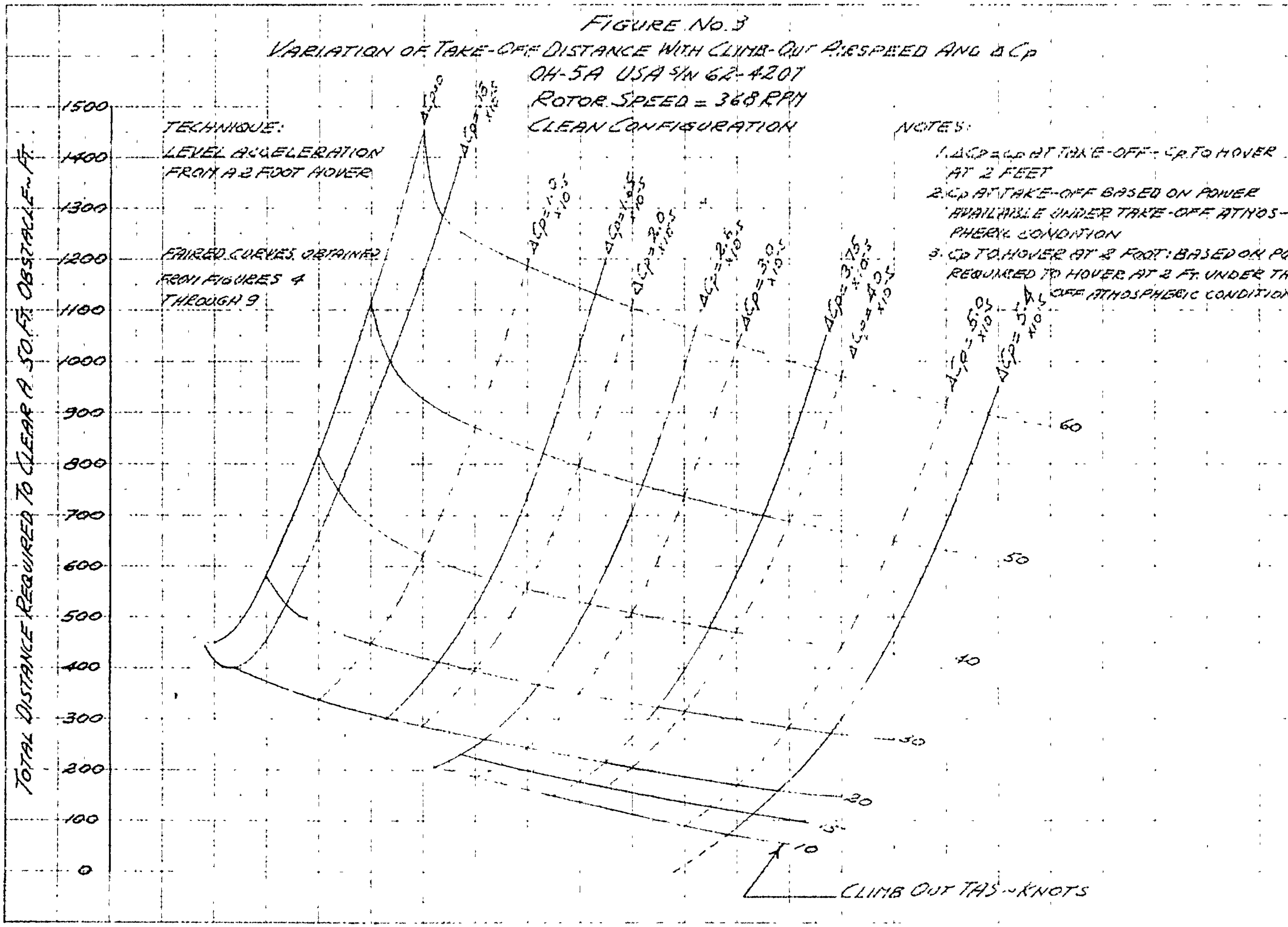


FIGURE No. 3
VARIATION OF TAKE-OFF DISTANCE WITH CLIMB-OUT AIRSPEED AND ΔCp
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
CLEAN CONFIGURATION
TECHNIQUE:
LEVEL ACCELERATION FROM A 2 FOOT HOVER
FAIRED CURVES OBTAINED FROM FIGURES 4 THROUGH 9
NOTES:
1. ΔCp = Cp AT TAKE-OFF - Cp TO HOVER AT 2 FEET
2. Cp AT TAKE-OFF BASED ON POWER AVAILABLE UNDER TAKE-OFF ATMOSPHERIC CONDITION
3. Cp TO HOVER AT 2 FOOT: BASED ON POWER REQUIRED TO HOVER AT 2 FT. UNDER TAKE-OFF ATMOSPHERIC CONDITIONS
ΔCp=0
ΔCp=1.0 x10^-5
ΔCp=1.5 x10^-5
ΔCp=2.0 x10^-5
ΔCp=3.0 x10^-5
ΔCp=3.75 x10^-5
ΔCp=4.0 x10^-5
ΔCp=5.0 x10^-5
ΔCp=5.4 x10^-5
TOTAL DISTANCE REQUIRED TO CLEAR A 50 FT. OBSTACLE ~ FT.
1500
1400
1300
1200
1100
1000
900
800
700
600
500
400
300
200
100
0
60
50
40
30
20
15
10
CLIMB OUT TAS ~ KNOTS

FIGURE No. 4
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
G.W. = 2730 LB
FAT = 10°C
Hp = 9500 FT.
WIND. ≦ 2 KNOTS
ΔCp = 0
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER
CLEAN CONFIGURATION

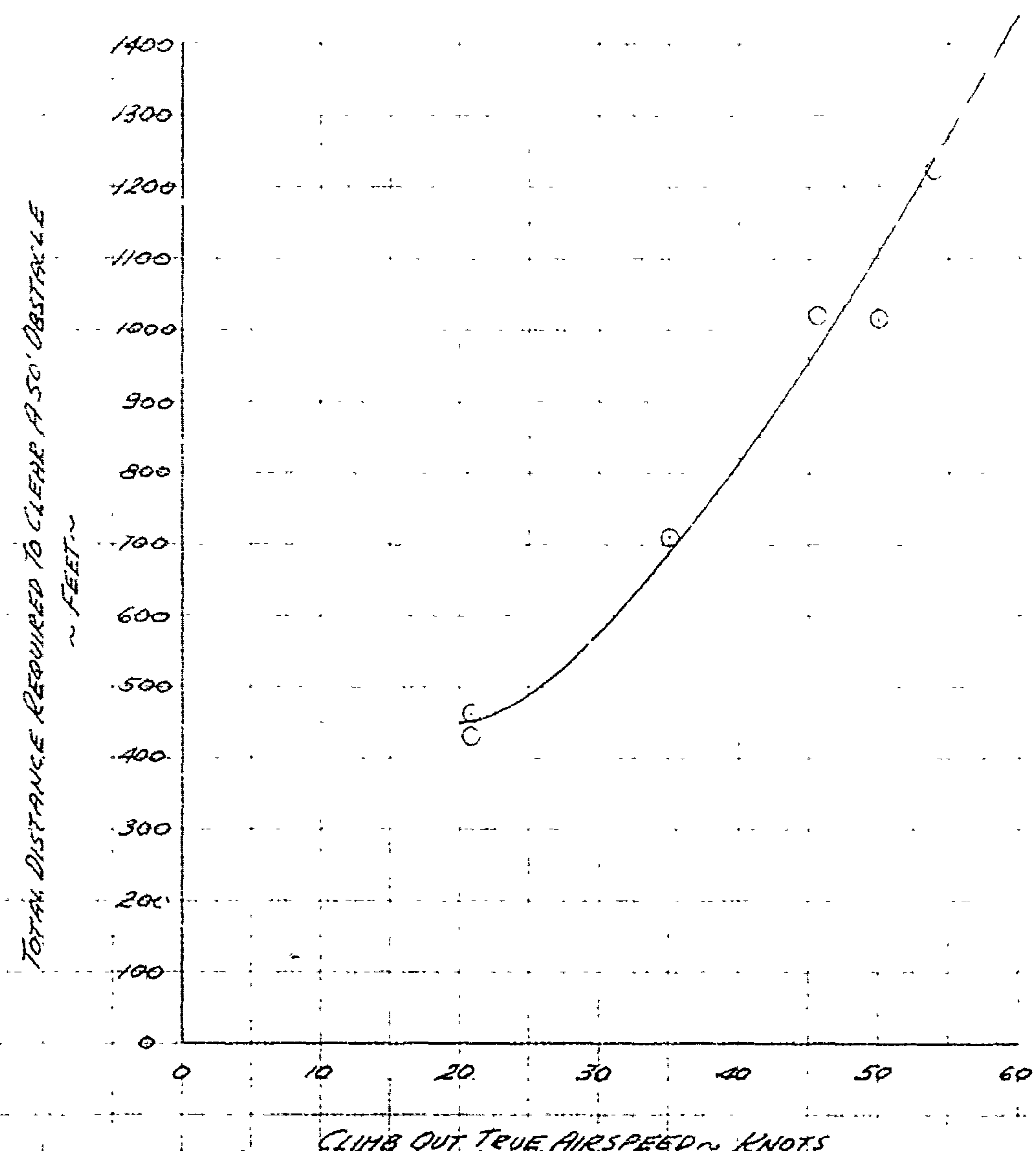


FIGURE No. 5
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
G.W. = 2540 LB.
FAT = 12 °C
Hp = 3400 FT.
WIND ≤ 2 KNOTS
$\Delta C_p = 0.18 \times 10^{-5}$
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
CLEAN CONFIGURATION

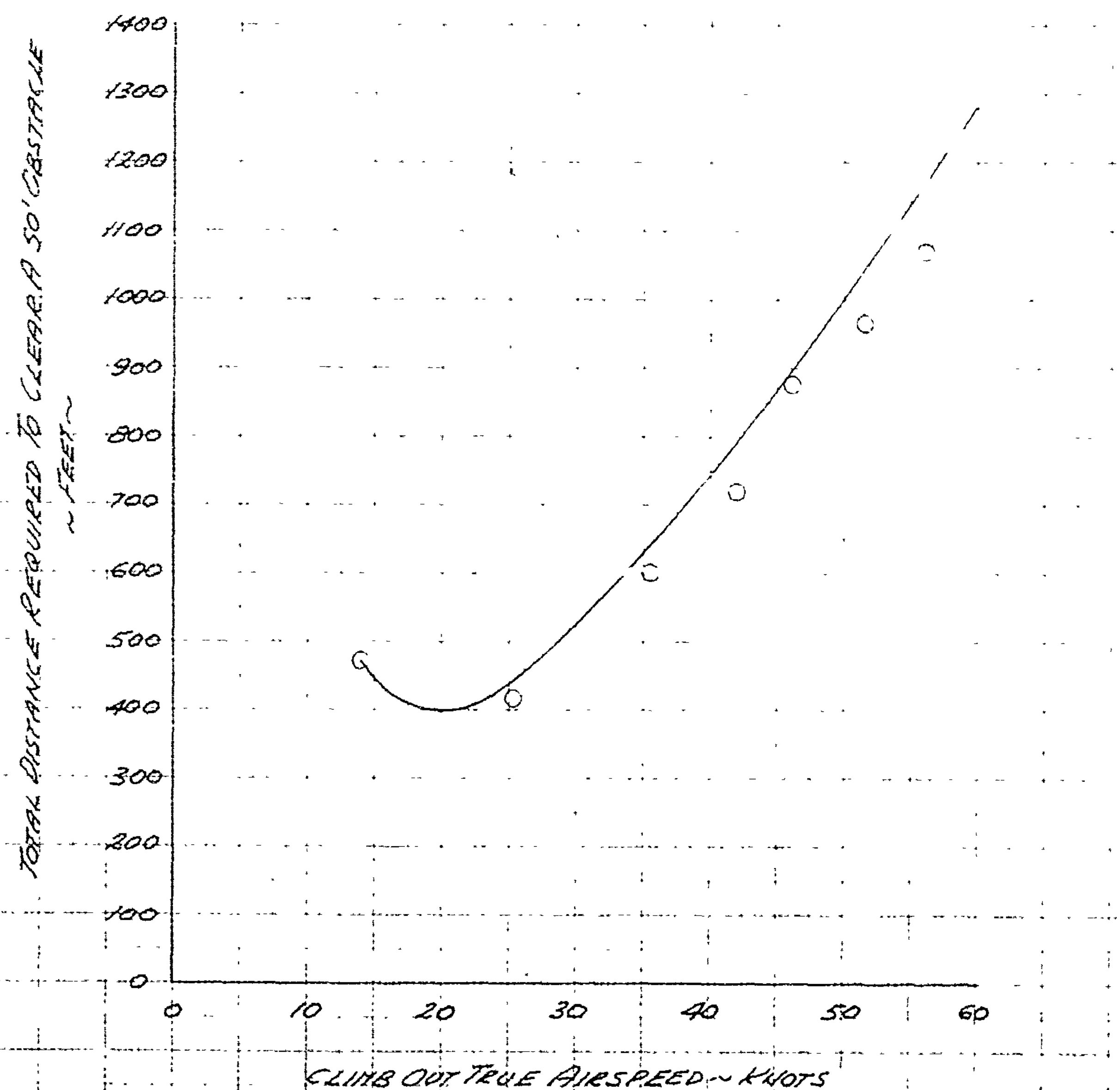
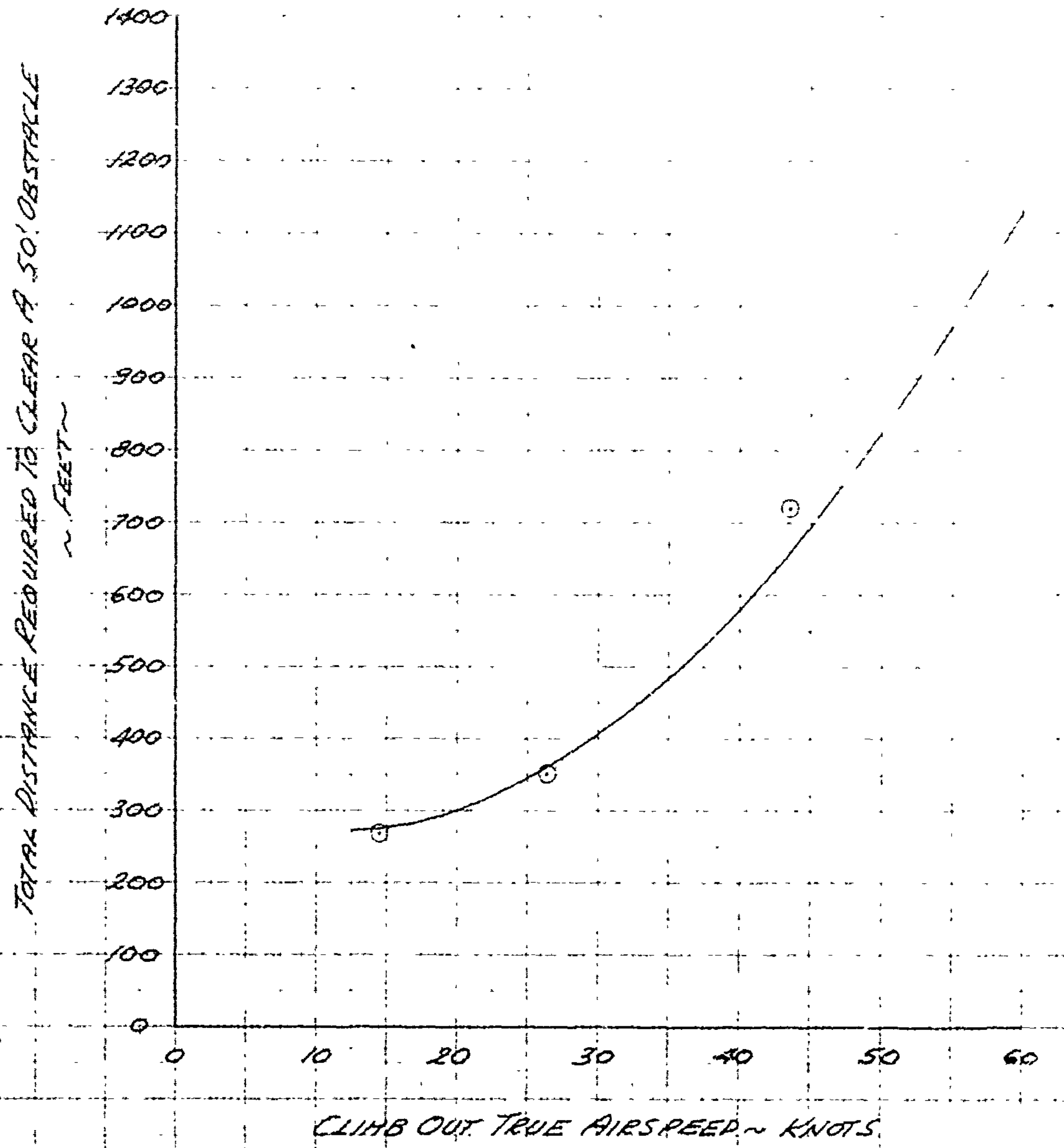


FIGURE NO. 6
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
G.W. = 2530 LB.
FAT = 10°C
Hp = 9535 FT.
WIND ≦ 2 KNOTS
$\Delta C_p = 1.65 \times 10^{-5}$
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE No. 7
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GW = 2950 LB.
FAT = 20°C
Hp = 3960 FT.
WIND ≦ 2 KNOTS
$\Delta C_F = 2.6 \times 10^{-5}$
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER
CLEAN CONFIGURATION

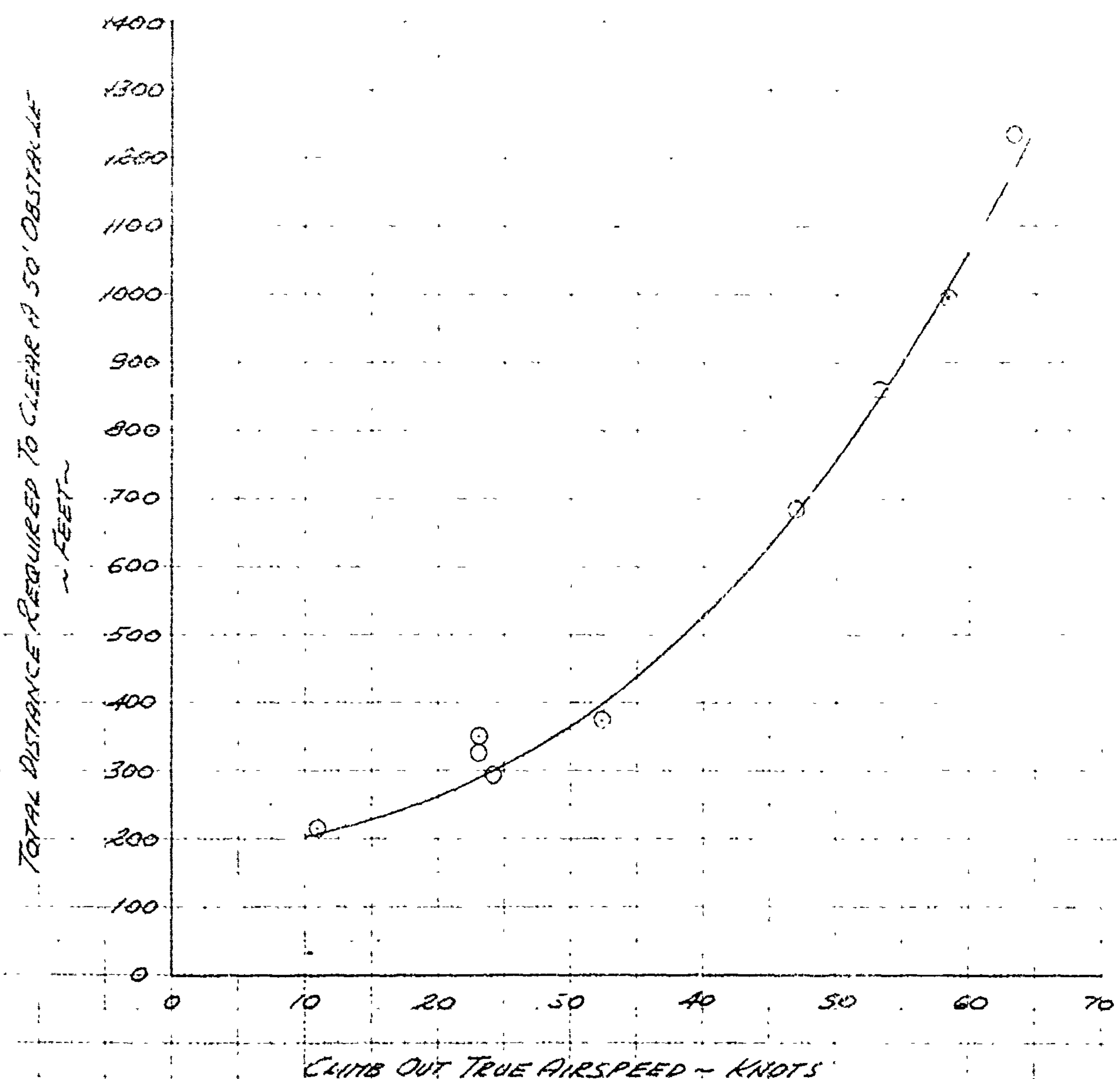
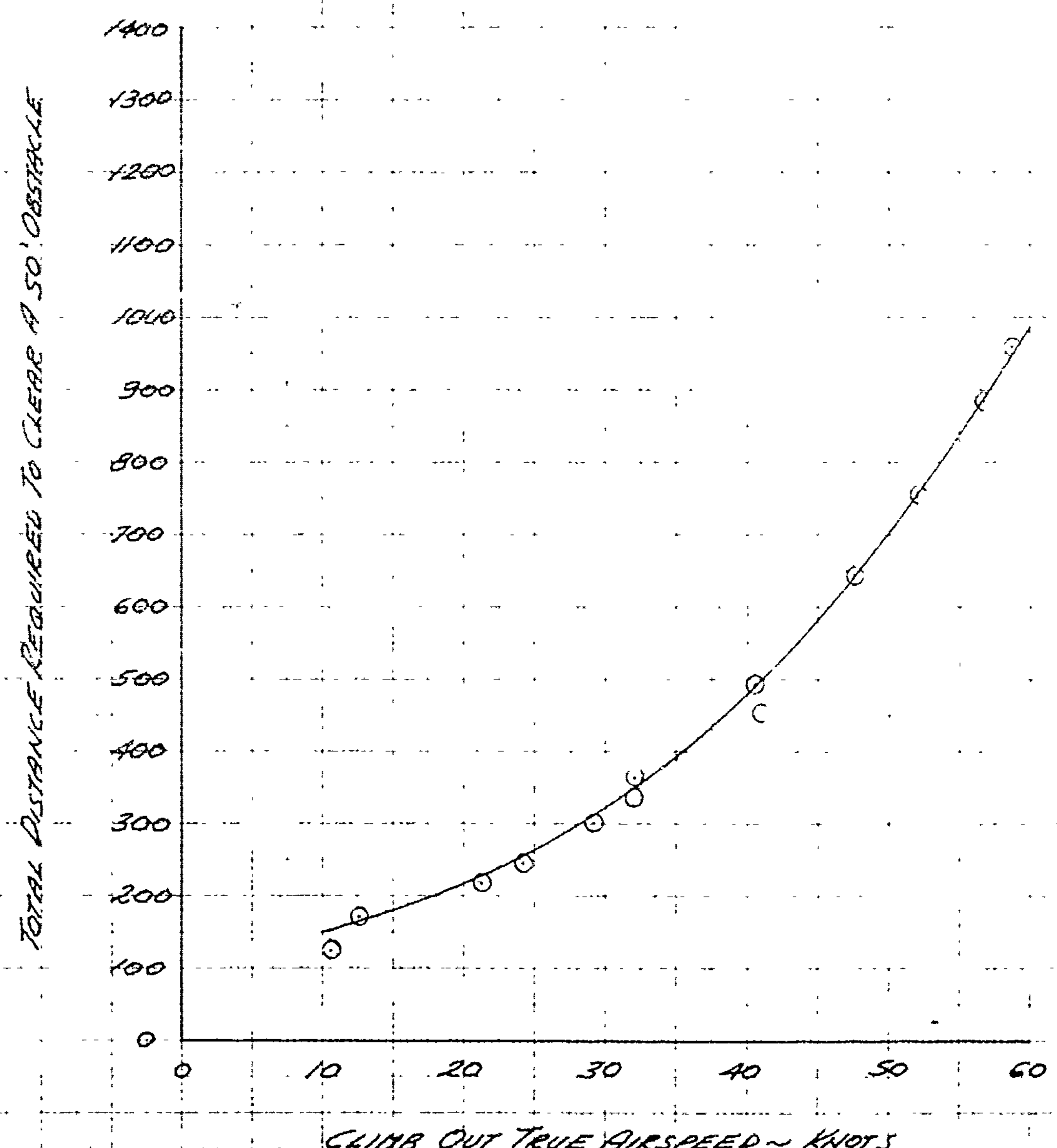


FIGURE NO. 8
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
G.W. = 2370 LB.
FAT = 27 °C
Hp = 4030 FT.
WIND ≦ 2 KNOTS
$\Delta C_p = 3.75 \times 10^{-5}$
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE No. 9
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
G.W. = 2540 LB.
FAT = 24°C
Hp = 5900 FT.
WIND ≦ 2 KNOTS
$\Delta C_p = 5.4 \times 10^{-5}$
TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER
CLEAN CONFIGURATION

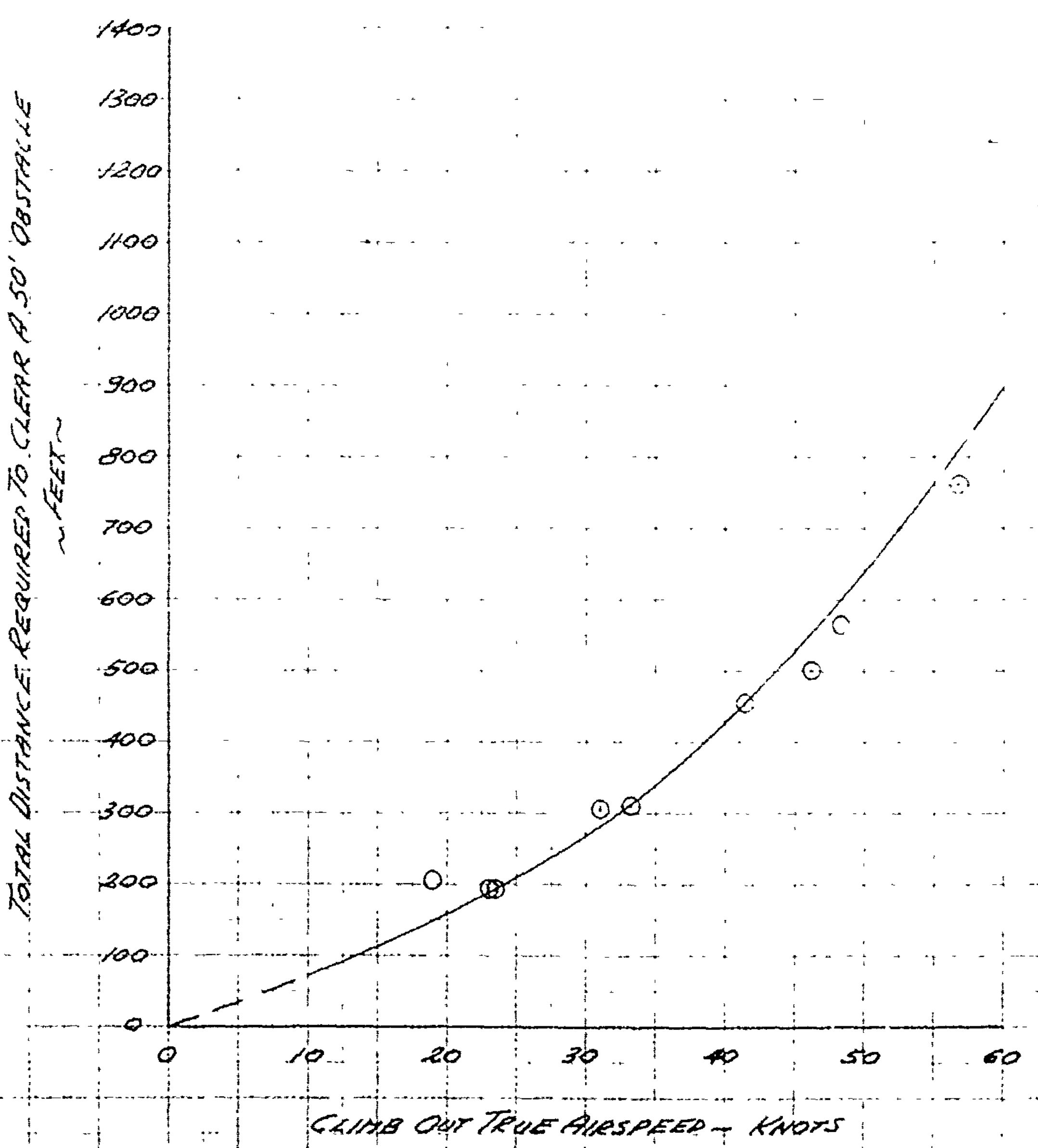
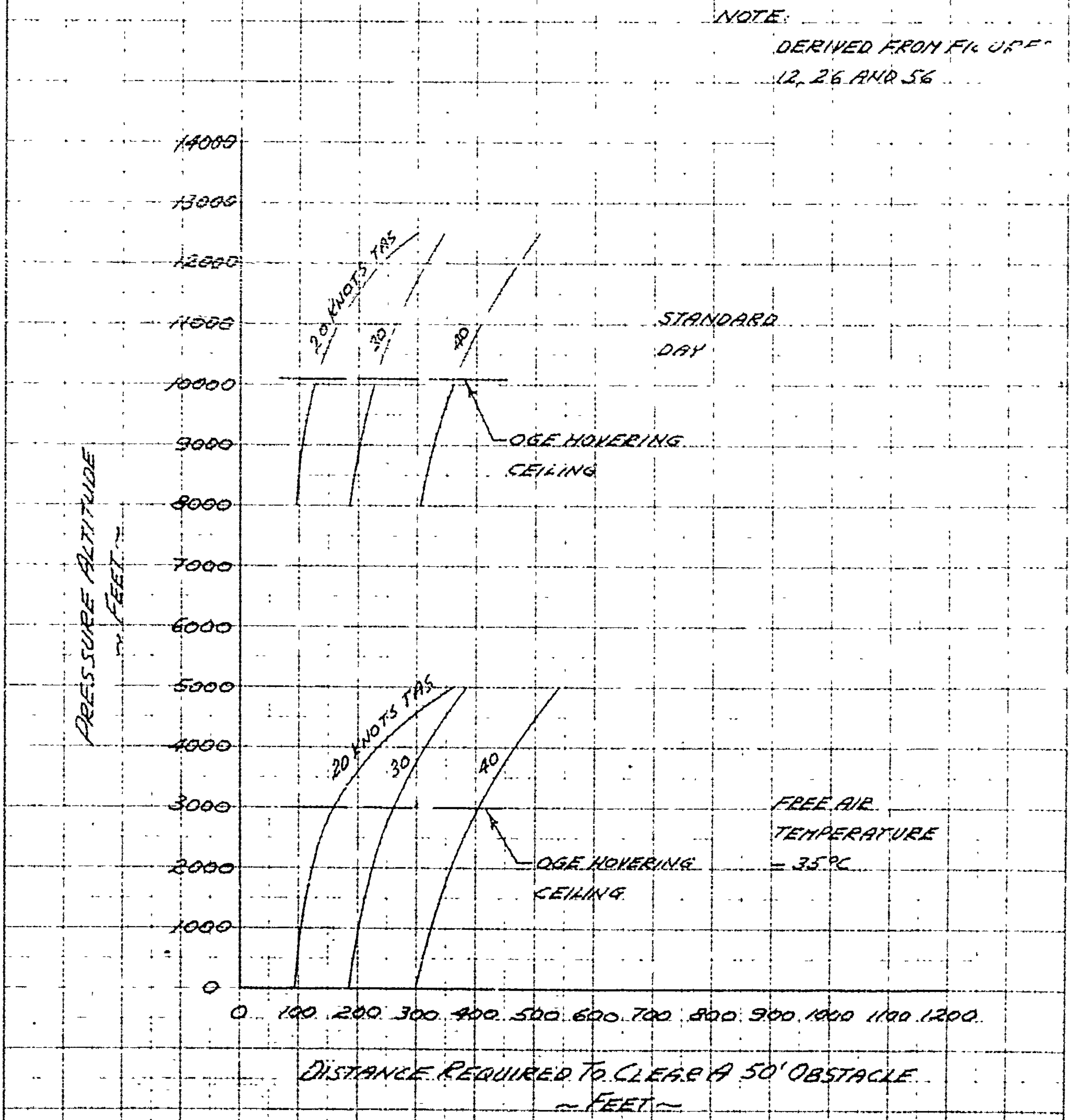


FIGURE NO. 10
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
TECHNIQUE: CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIGHT ON SKIDS
GROSS WEIGHT = 2530 LB
ROTOR SPEED = 368 RPM
CLEAN CONFIGURATION
TAKE-OFF POWER



FOR OFFICIAL USE ONLY

FIGURE No 11.
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
TECHNIQUE: CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIGHT ON SKIDS
GROSS WEIGHT = 3000 LB.
ROTOR SPEED = 368 RPM
CLEAN CONFIGURATION
TAKE-OFF POWER.

NOTE:
DERIVED FROM FIGURE 12, 26 AND 56

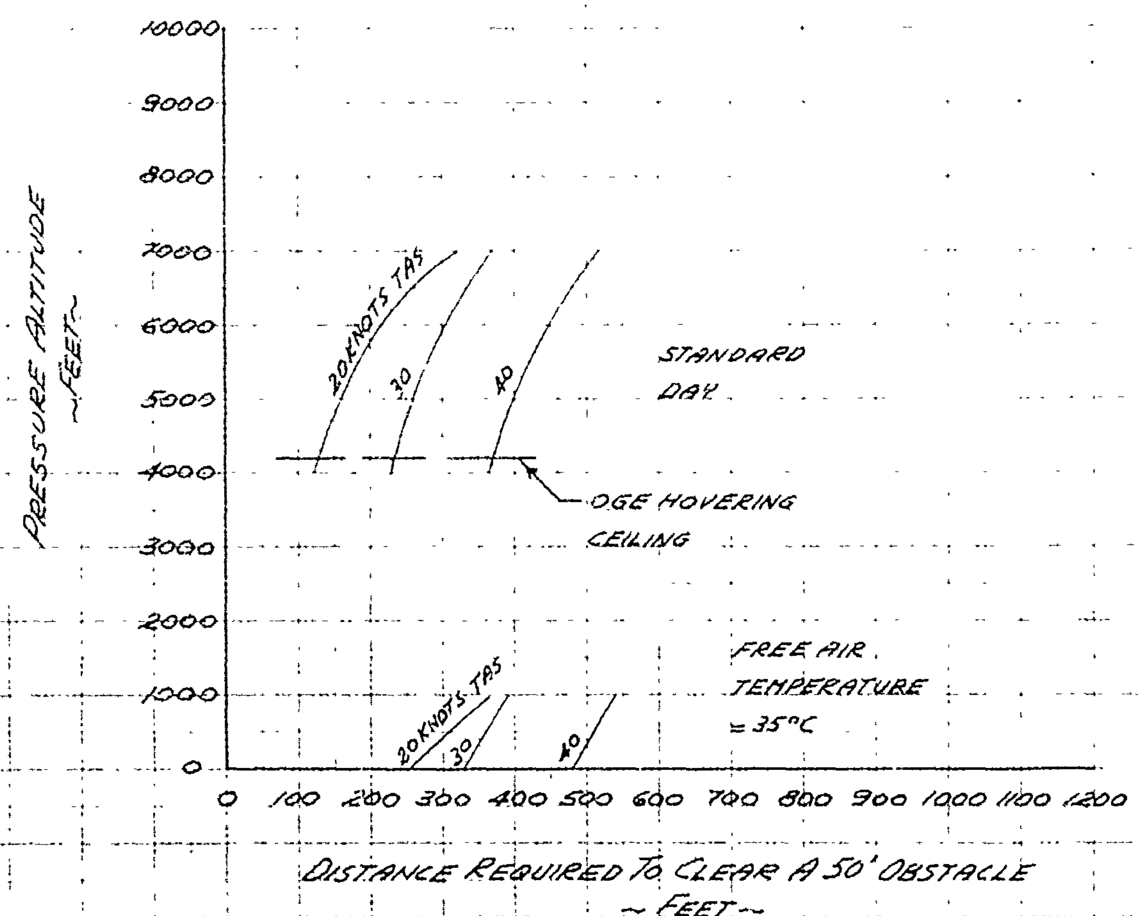


FIGURE NO. 12
VARIATION OF TAKE-OFF DISTANCE WITH CLIMB-OUT AIRSPEED AND ΔCp
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
CLEAN CONFIGURATION

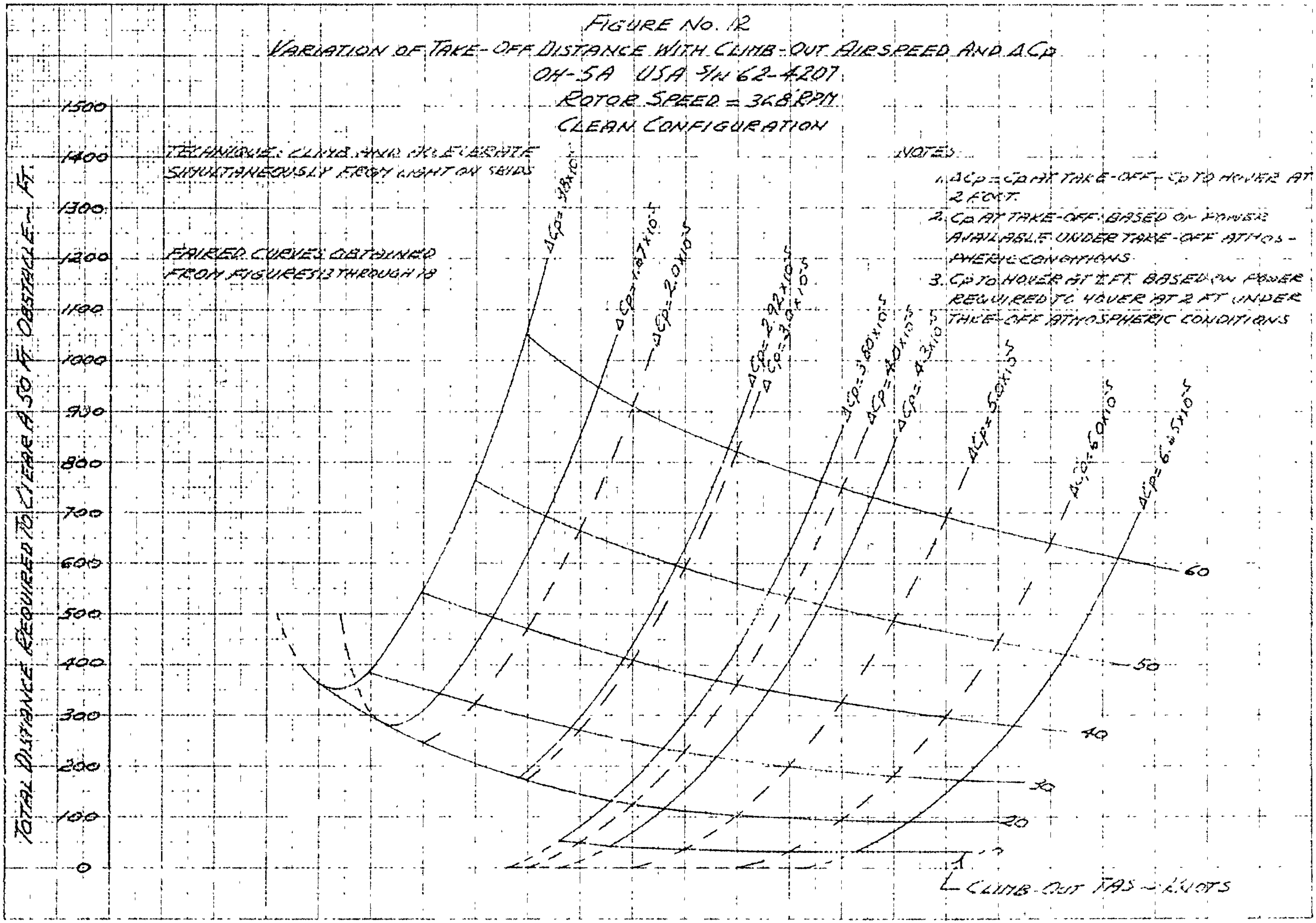
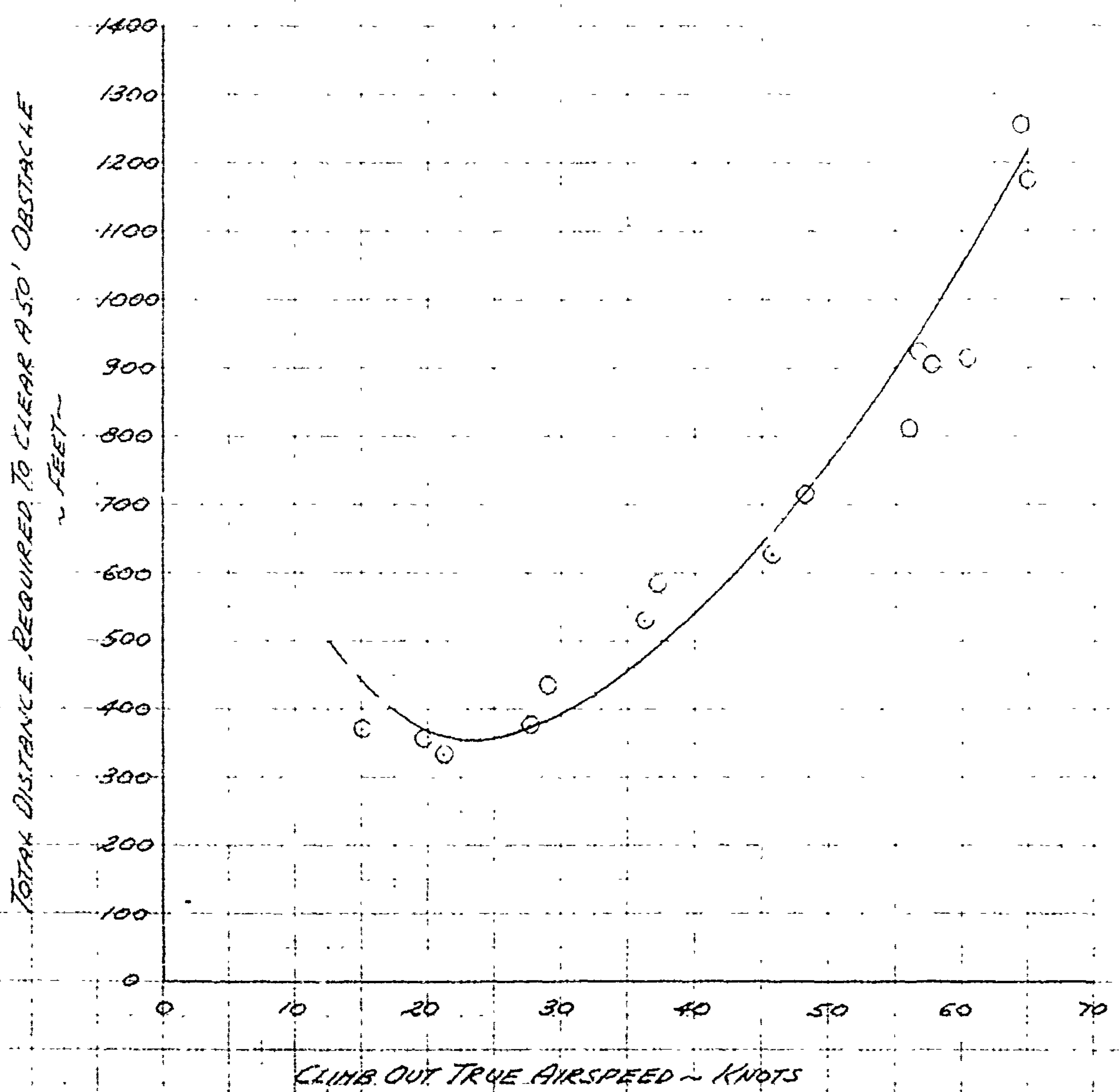


FIGURE No. 13
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH 5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GROSS WEIGHT = 2430 LB.
FAT = 5°C
H_D = 9560 FT.
WIND ≦ 2 KNOTS
$\Delta C_P = 0.98 \times 10^{-5}$
TECHNIQUE: CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIGHT ON SKIDS
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE NO. 14

TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GROSS WEIGHT = 2520 LB.
FAT = 22 °C
Hp = 4180 FT.
WIND ≅ 2 KNOTS
$\Delta C_P = 1.67 \times 10^{-5}$
TECHNIQUE: CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIGHT ON SKIDS
CLEAN CONFIGURATION

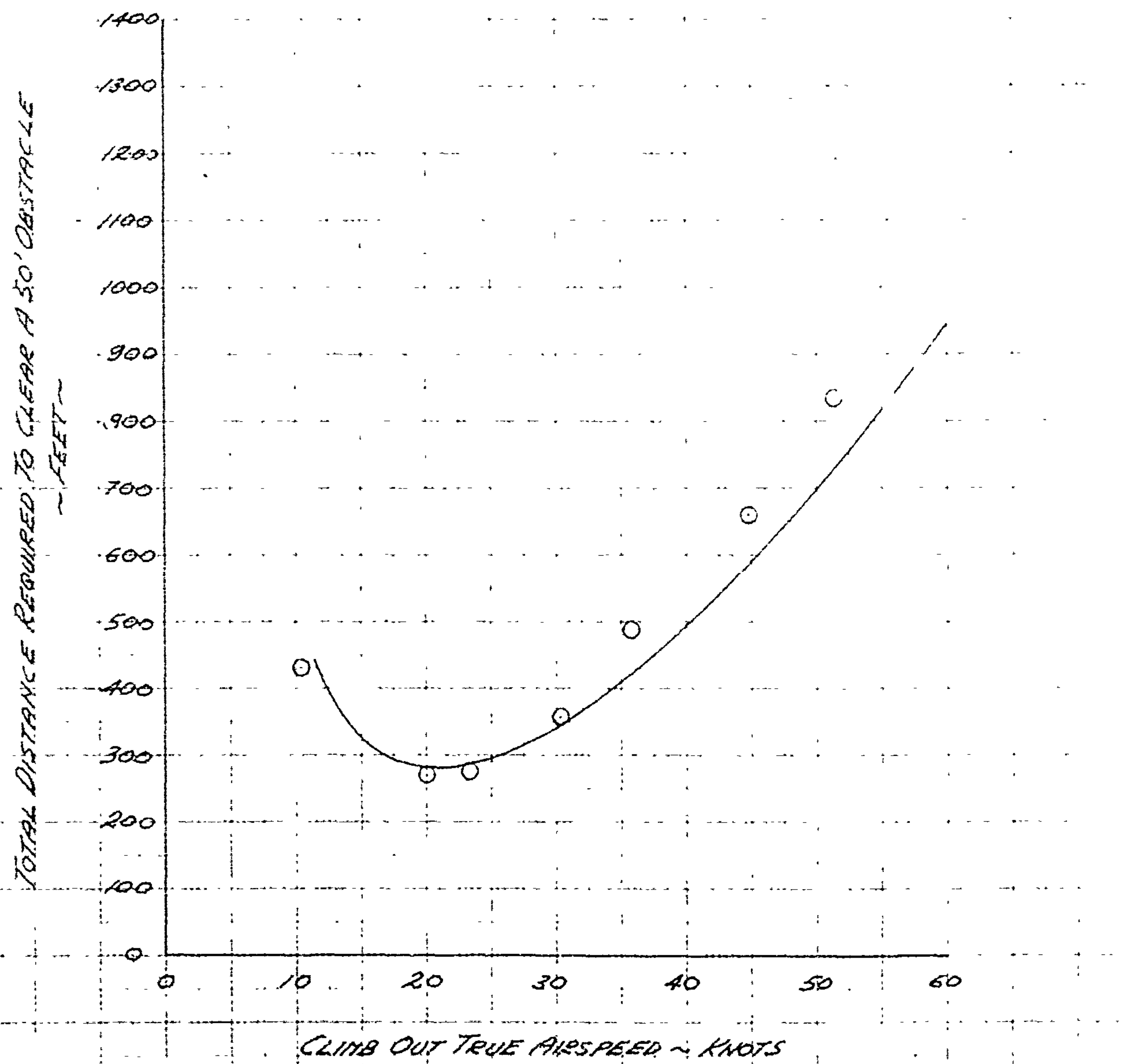


FIGURE No. 15
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GROSS WEIGHT = 2330 LB.
FAT = 8°C
Hp = 9560 FT.
WIND ≥ 2 KNOTS
$\Delta C_P = 2.92 \times 10^{-5}$
TECHNIQUE: CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIGHT ON SKIDS
CLEAN CONFIGURATION

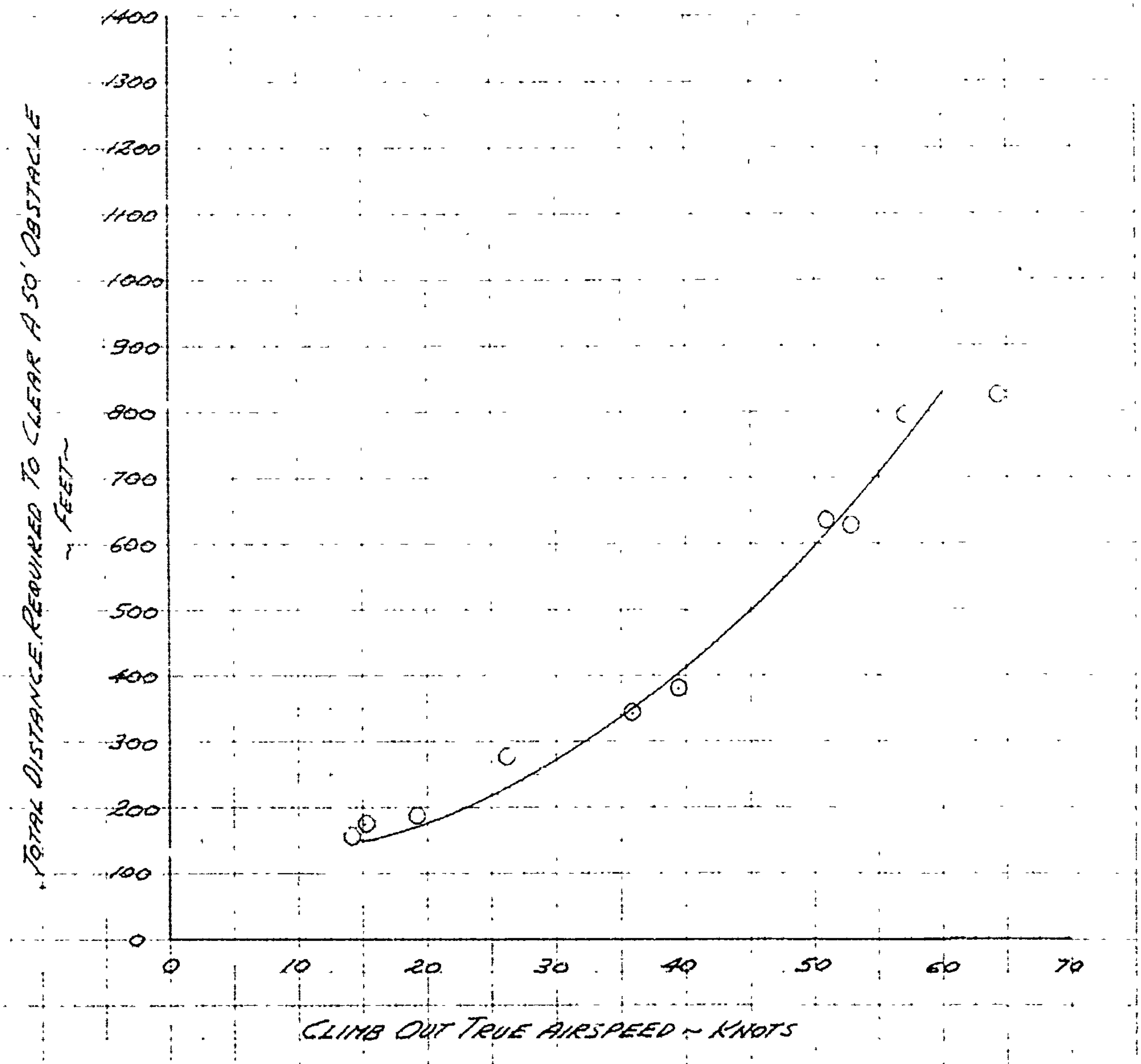


FIGURE NO. 16
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GROSS WEIGHT = 2920 LB.
FAT = 8.5°C
H_P = 4200 FT.
WIND ≦ 2 KNOTS
$\Delta C_P = 3.8 \times 10^{-5}$
TECHNIQUE · CLIMB AND ACCELERATE
SIMULTANEOUSLY FROM LIGHT ON SKIDS
CLEAN CONFIGURATION

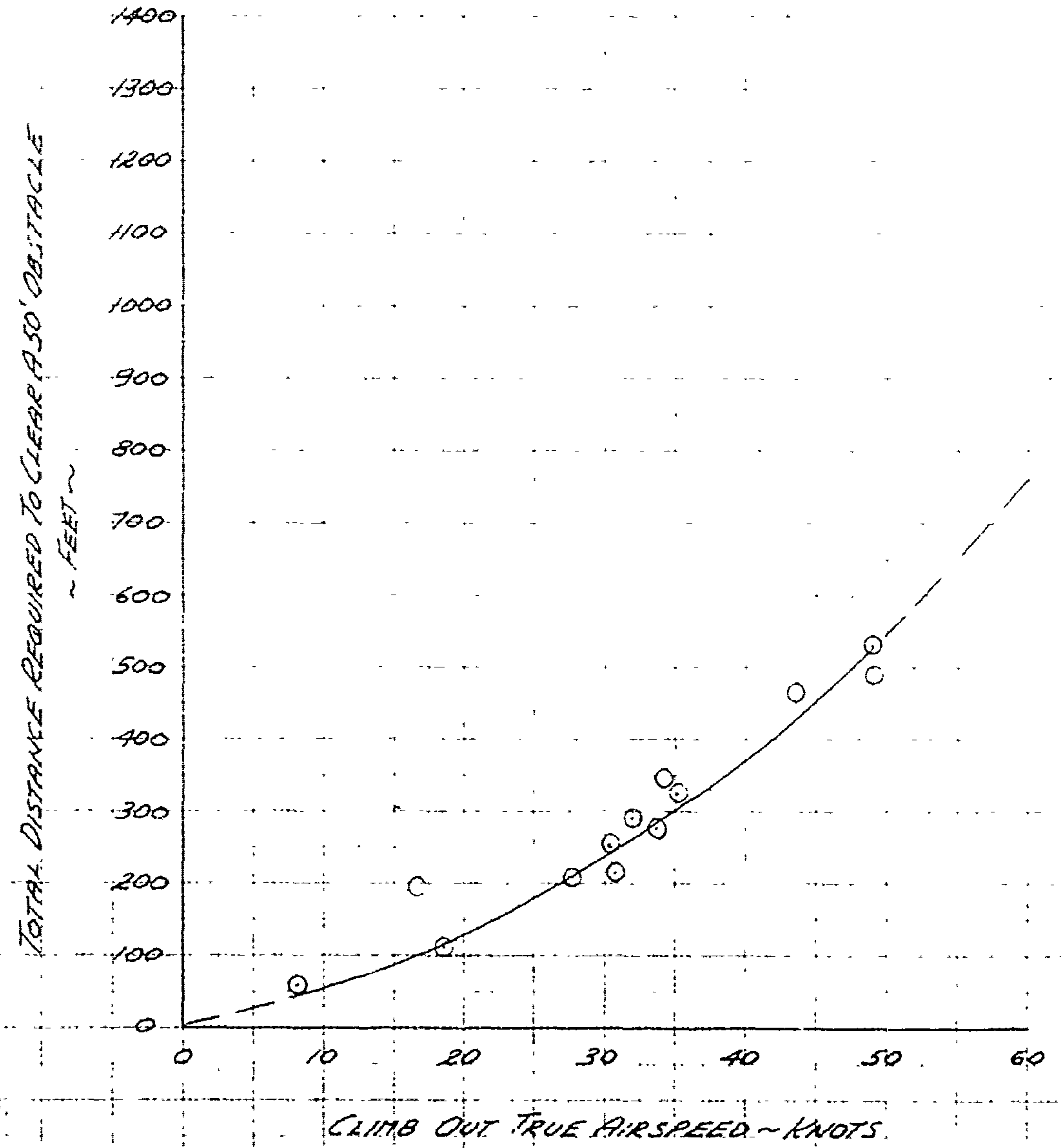


FIGURE No. 17
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GROSS WEIGHT = 2540 LB.
F.A.T. = 23.4°C
Hp = 4080 FT
WIND ≦ 2 KNOTS
$\Delta C_P = 4.3 \times 10^{-5}$
TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS
CLEAN CONFIGURATION

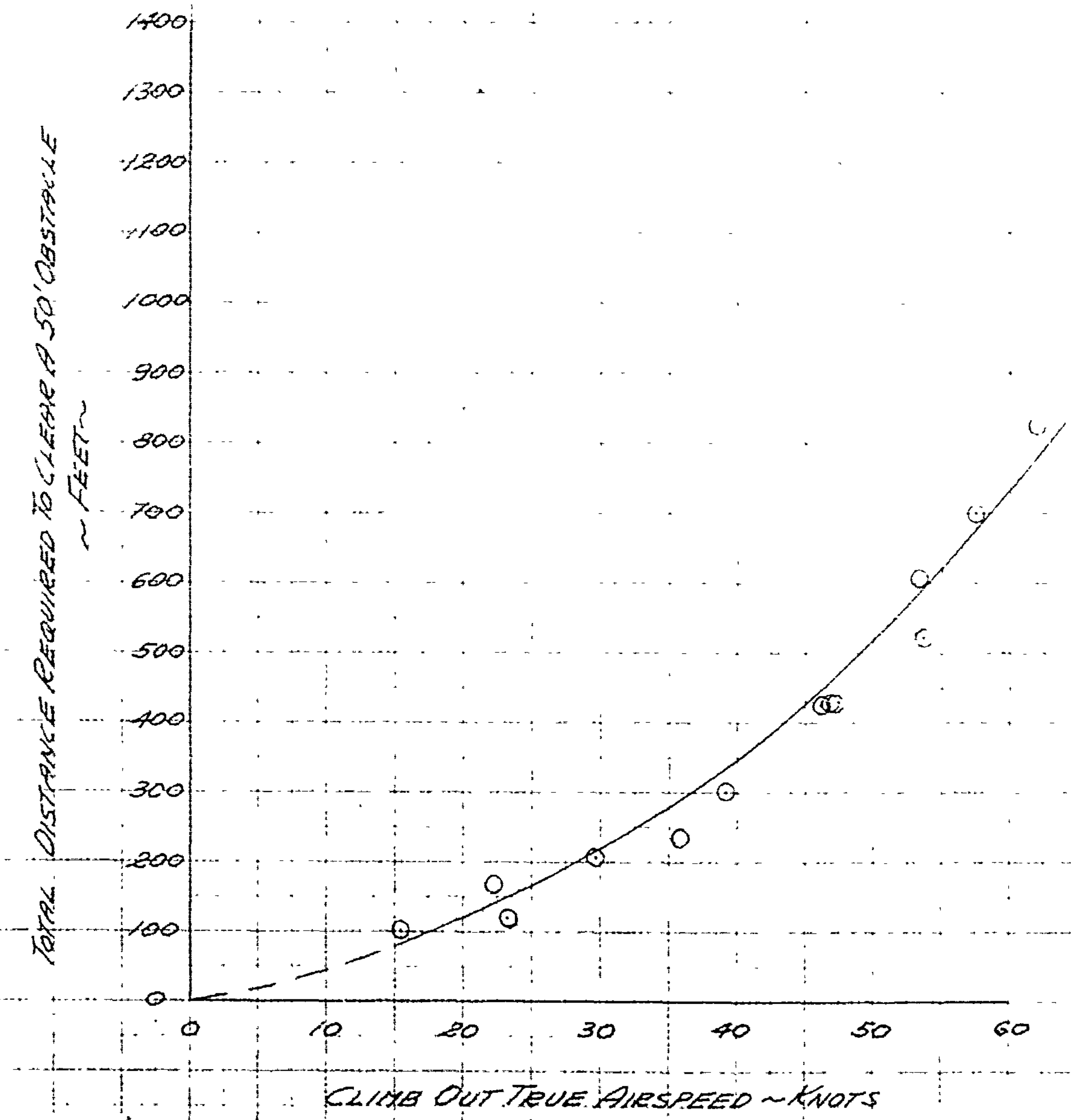


FIGURE No. 18
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GROSS WEIGHT = 2550 LB.
FAT = 9.5°C
Hp = 4250 FT.
WIND ≦ 2 KNOTS
$\Delta C_p = 6.65 \times 10^{-5}$
TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS
CLEAN CONFIGURATION

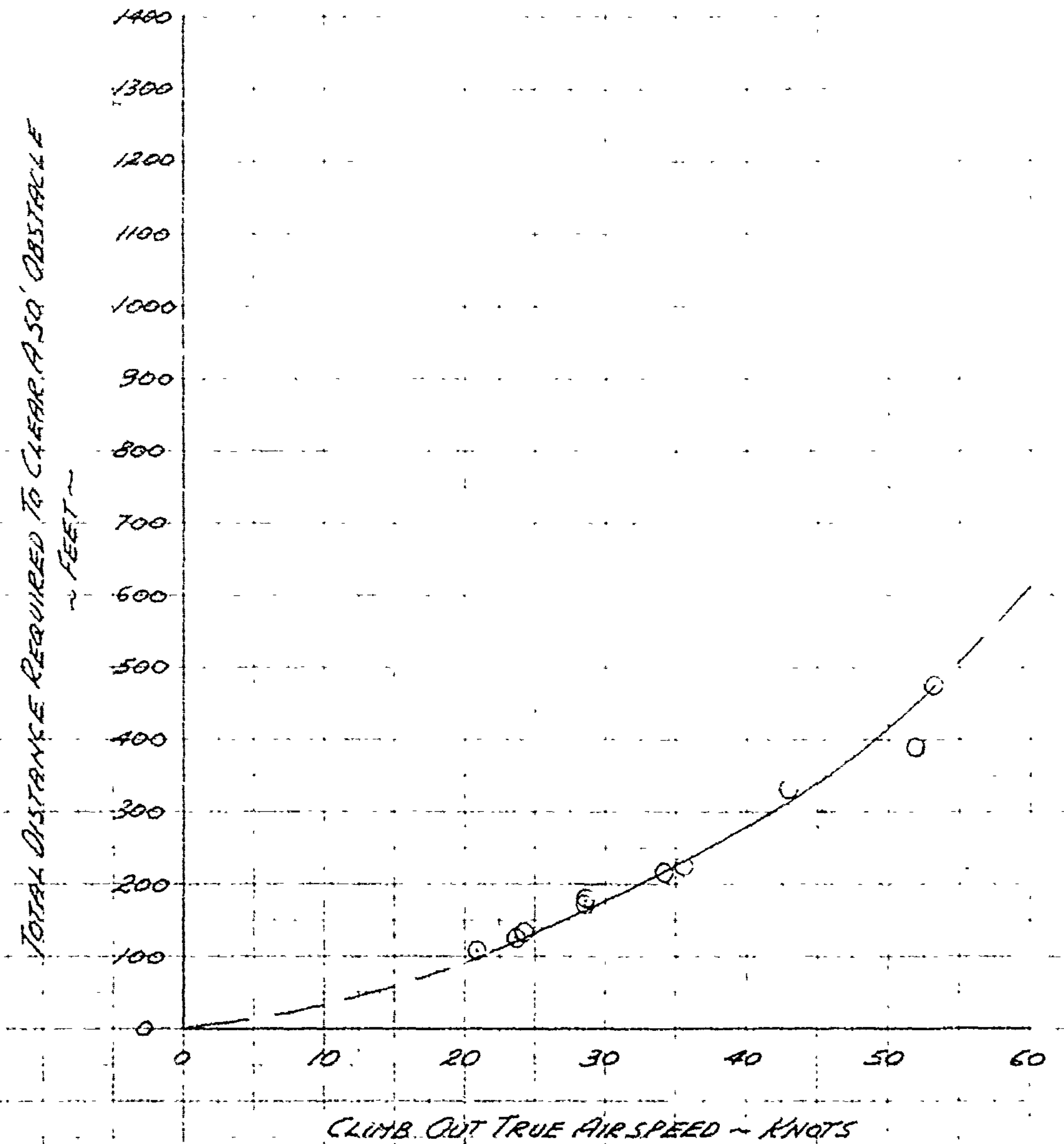
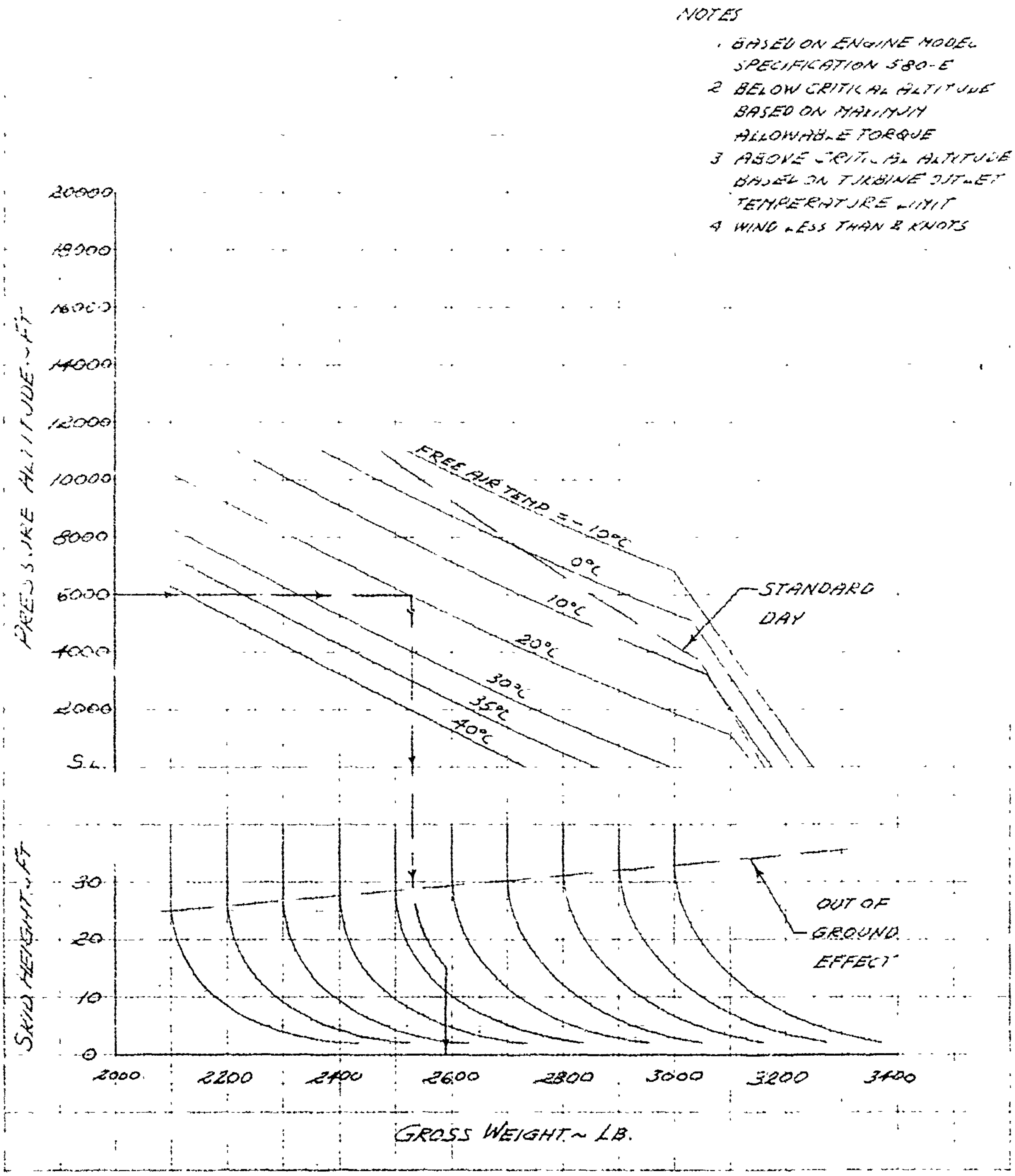


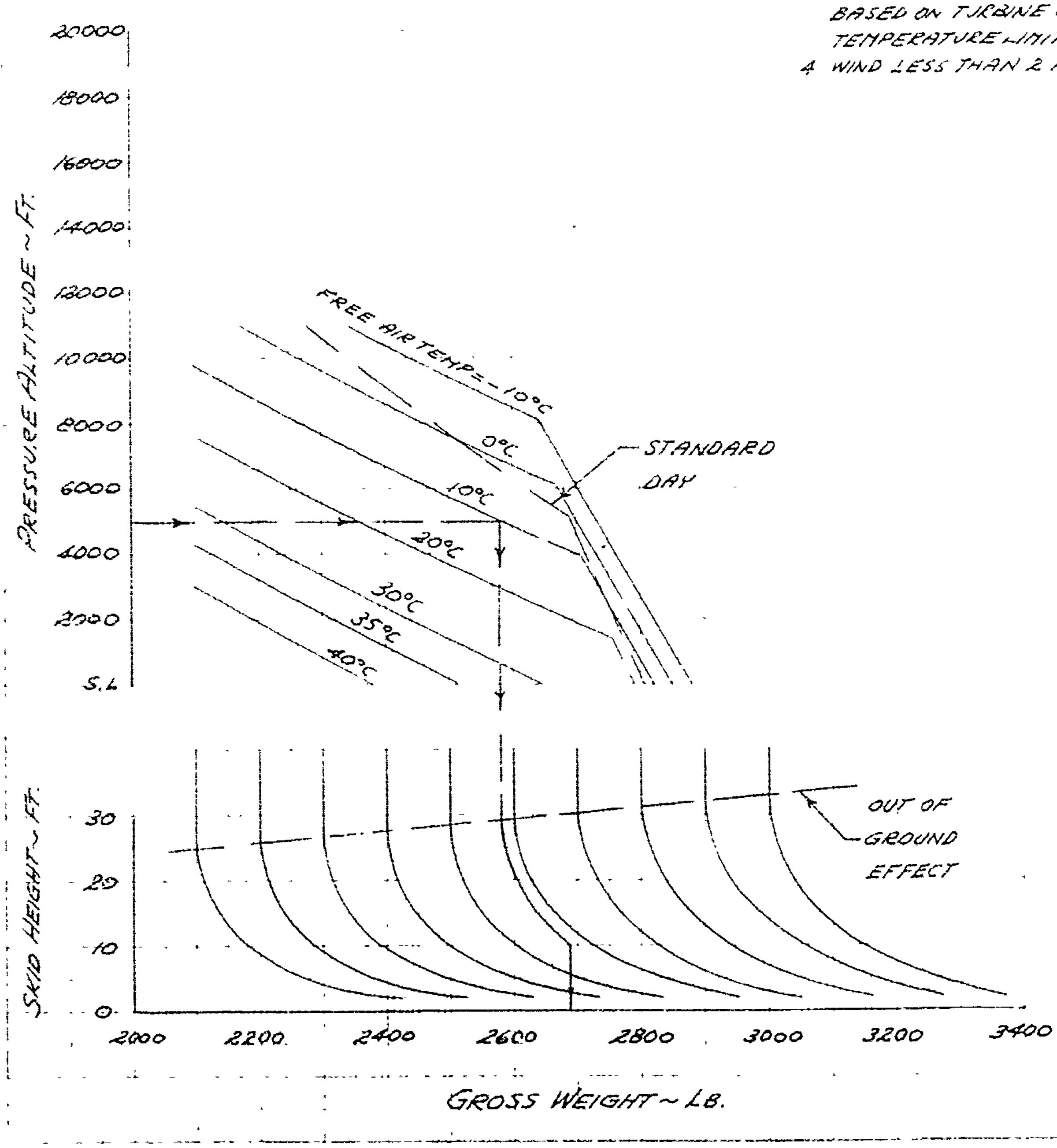
FIGURE NO 19
SUMMARY HOVERING PERFORMANCE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
TAKE-OFF POWER
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE NO. 20
SUMMARY HOVERING PERFORMANCE
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
MAXIMUM CONTINUOUS POWER
CLEAN CONFIGURATION

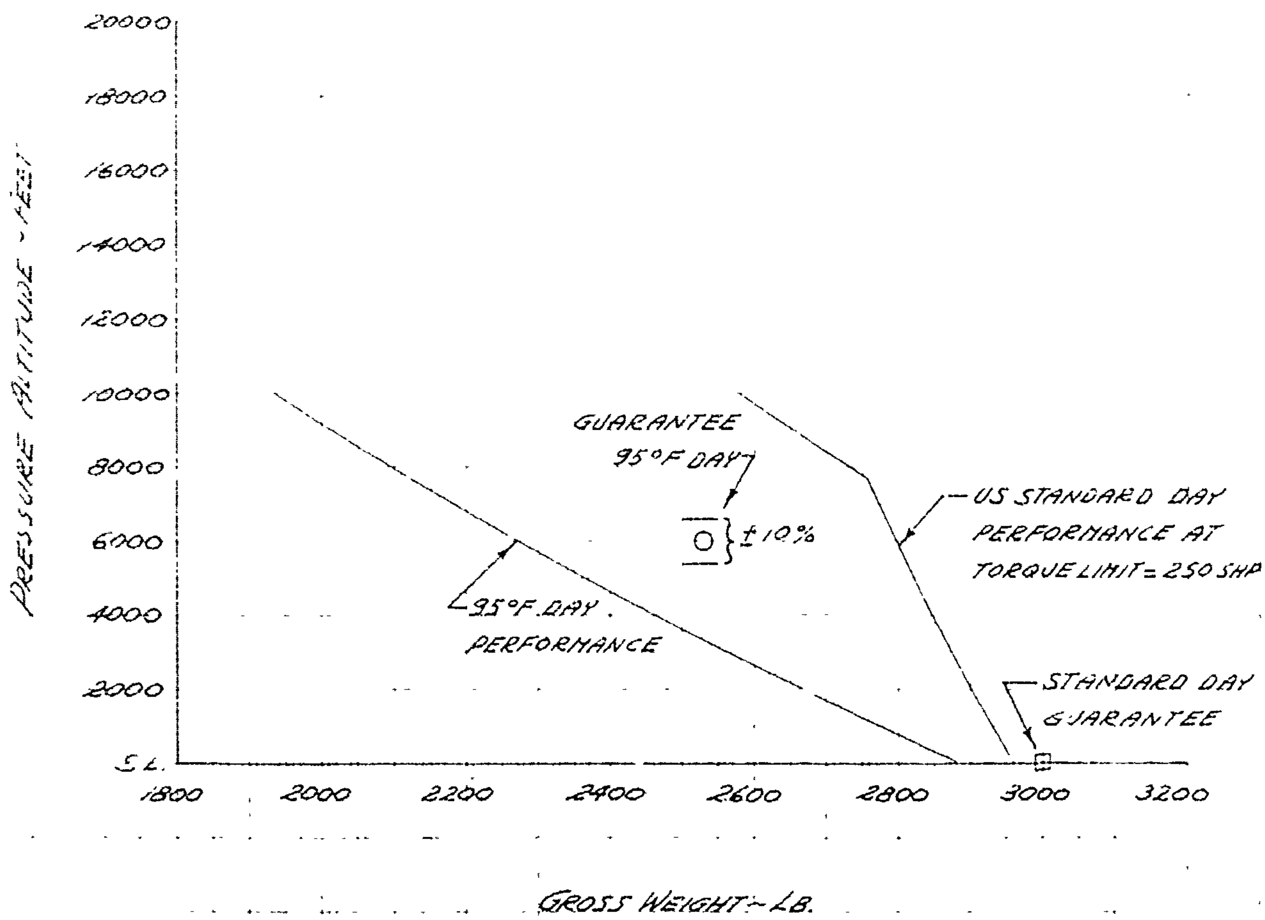
NOTES
1. BASED ON ENGINE MODEL SPECIFICATION 580-E
2. BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE
3. ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT
4. WIND LESS THAN 2 KNOTS



FOR OFFICIAL USE ONLY

FIGURE No 21
SUMMARY HOVERING PERFORMANCE
OH-5A USA S/N 62-4201
OUT OF GROUND EFFECT
ROTOR SPEED = 368 RPM
TAKE-OFF POWER

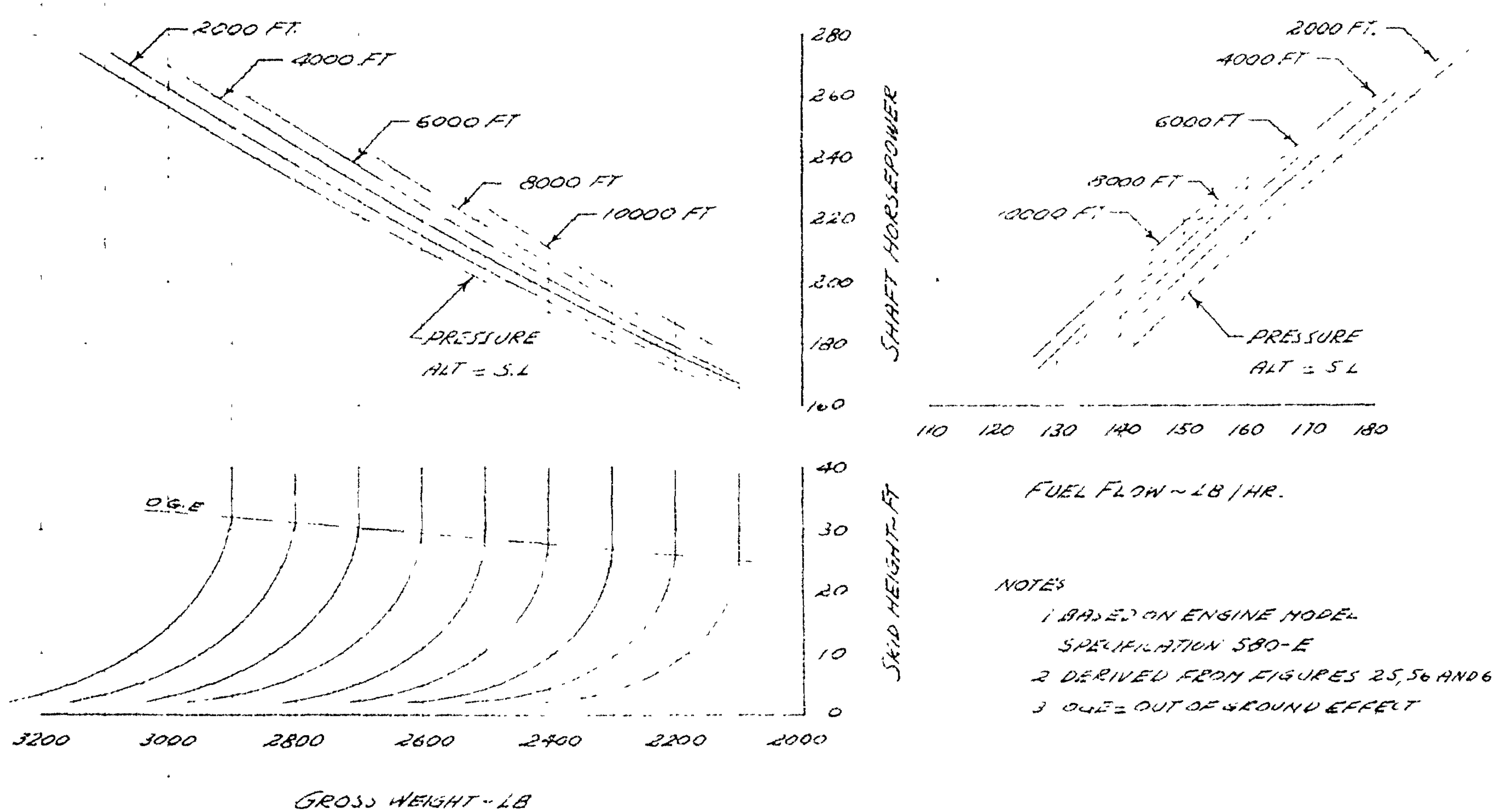
NOTES
1 BASED ON ENGINE MODEL SPECIFICATION 580-A
2 BELOW CRITICAL ALTITUDE BASED ON MAXIMUM ALLOWABLE TORQUE
3 ABOVE CRITICAL ALTITUDE BASED ON TURBINE OUTLET TEMPERATURE LIMIT



FOR OFFICIAL USE ONLY

FIGURE NO 22
FUEL FLOW DURING HOVERING FLIGHT
OH-5A USA S/N 62-42[illegible]

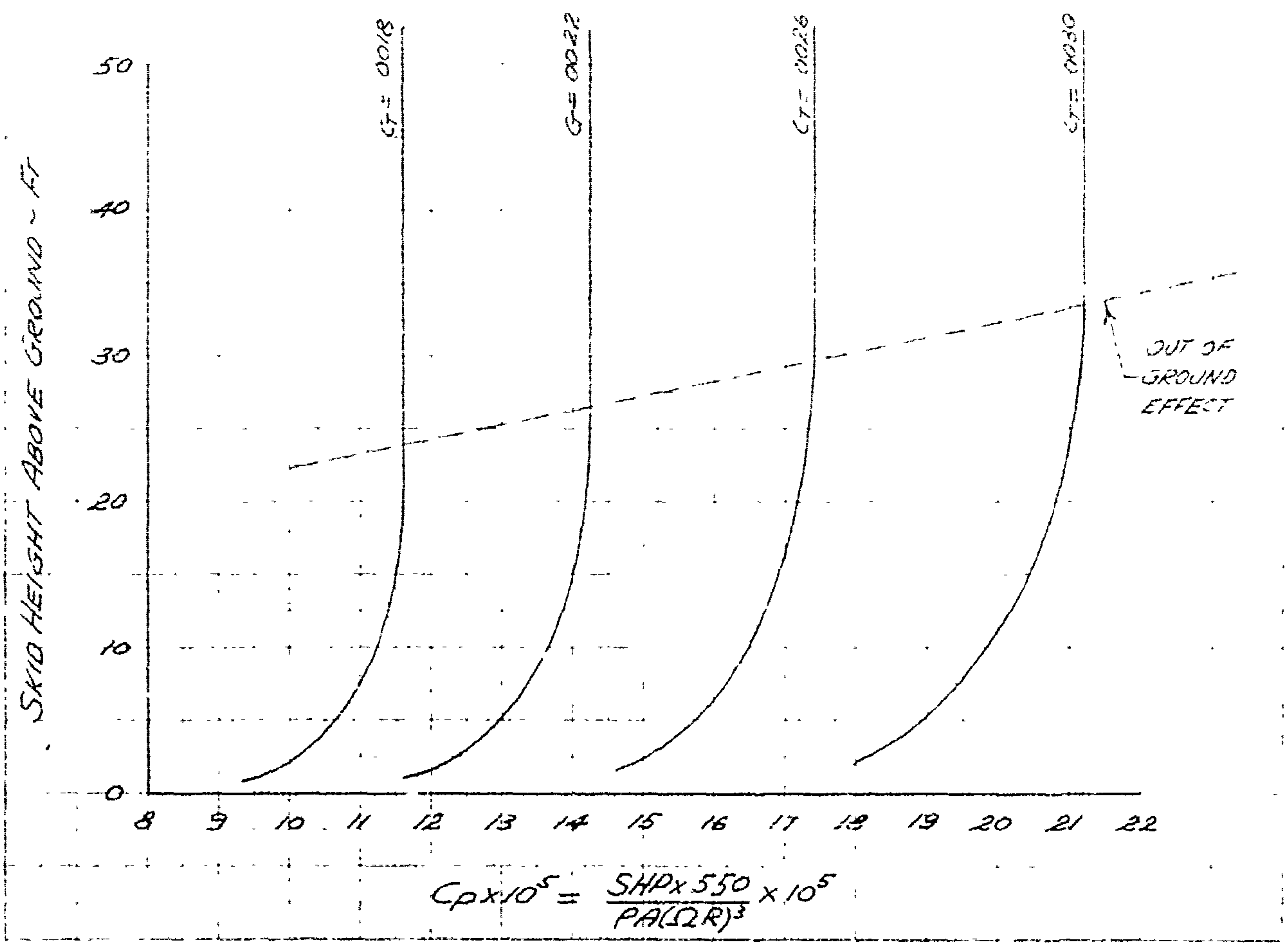
US STANDARD DAY



NOTES
1 BASED ON ENGINE MODEL SPECIFICATION 580-E
2 DERIVED FROM FIGURES 25, 56 AND 61
3 OGE = OUT OF GROUND EFFECT

FIGURE No. 23
HOVERING PERFORMANCE
OH-5A USA S/N 62-4207
FREE FLIGHT
CLEAN CONFIGURATION

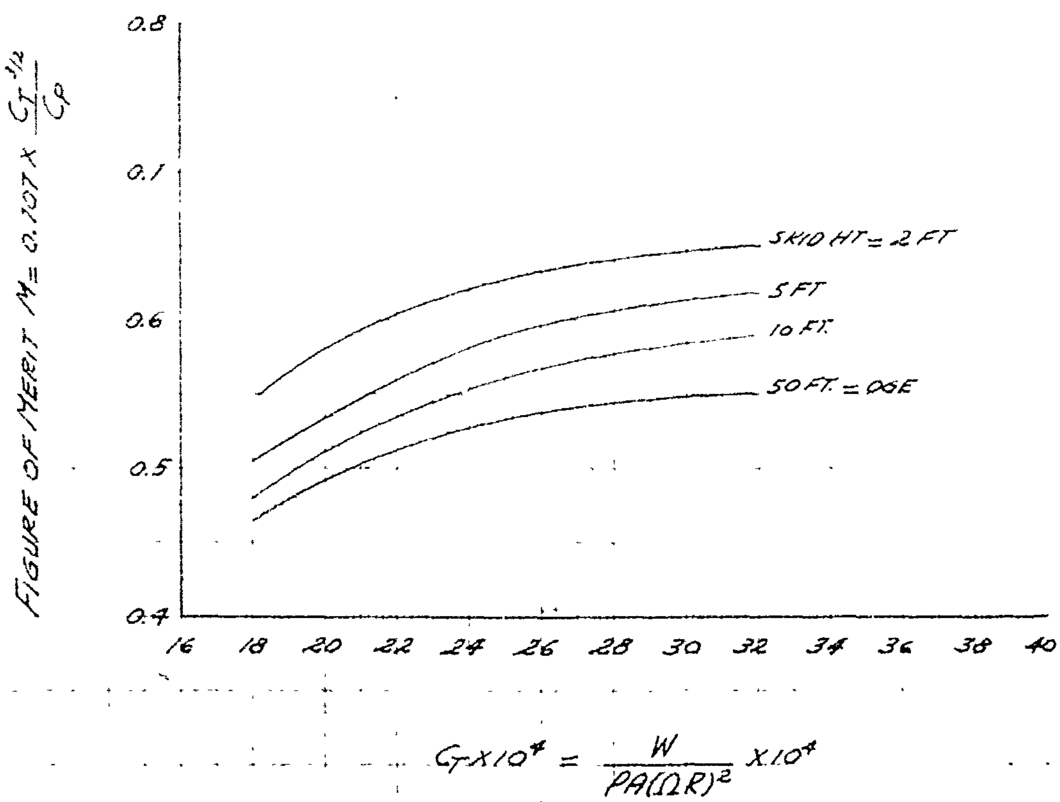
NOTES
1. DERIVED FROM FAIRED CURVES ON FIGURE 25
2. VERTICAL DISTANCE FROM BOTTOM OF SKIDS TO CENTER OF ROTOR HUB = 8.77 FEET



FOR OFFICIAL USE ONLY

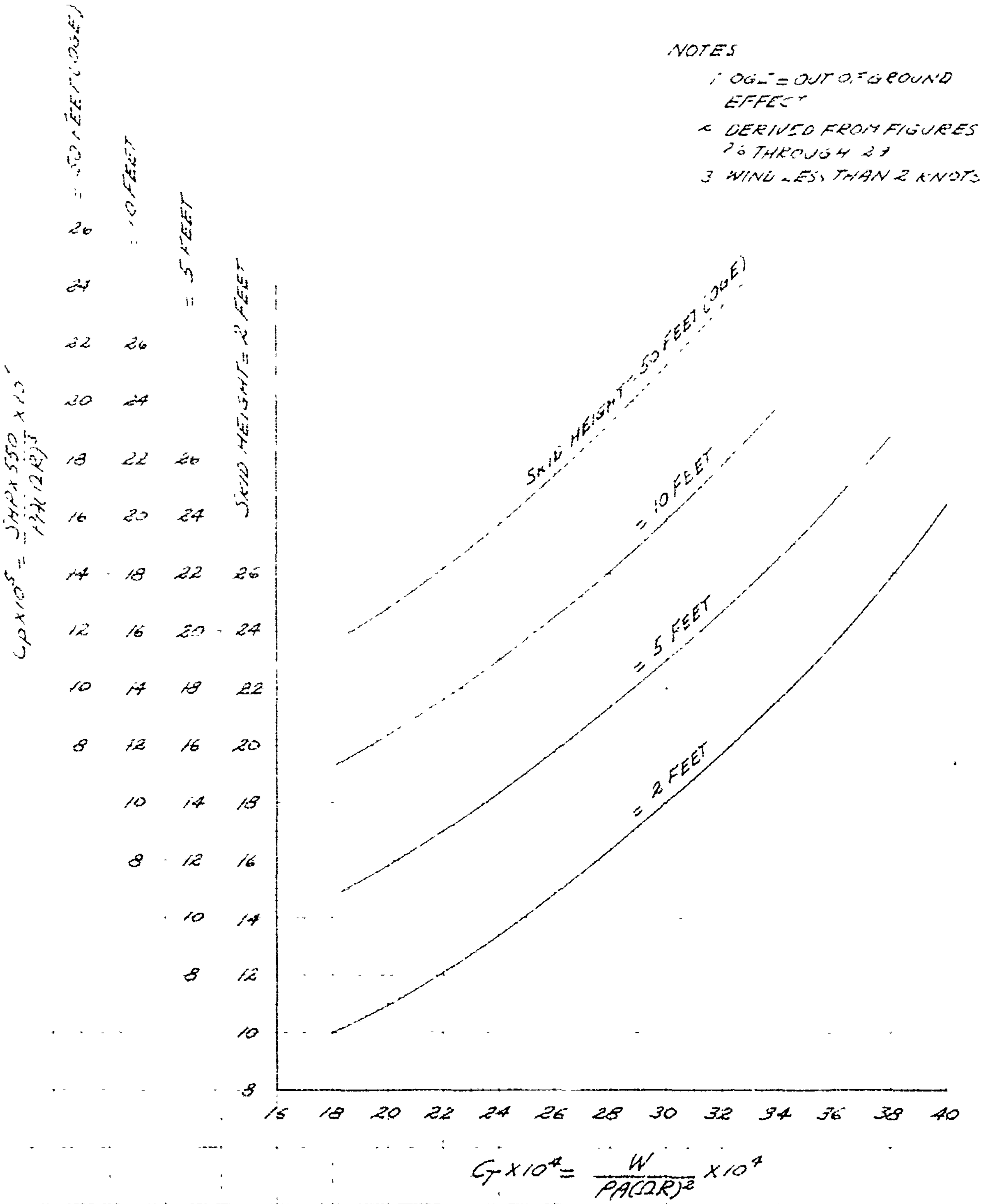
FIGURE No. 24
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-5A USA S/N 62-4201

NOTES
1. DERIVED FROM FIGURES 26 THROUGH 29
2. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.17 FEET
3. OGE = OUT OF GROUND EFFECT



FOR OFFICIAL USE ONLY

FIGURE NO. 25
HOVERING PERFORMANCE
OH-5A USA S/N 62-4207
FREE FLIGHT
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE NO. [illegible]
HOVERING PERFORMANCE
OH-5A USA S/N 62-4201
FREE FLIGHT
SKID HEIGHT = 2 FEET
CLEAN CONFIGURATION

ROTOR SPEED RPM			DENSITY ALT FEET
353	360	368	
○	⌀	○	S.L.
△	△	△	6360
□	□	□	10450

NOTES
1 VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.77 FEET
✓ WIND LESS THAN 2 KNOTS

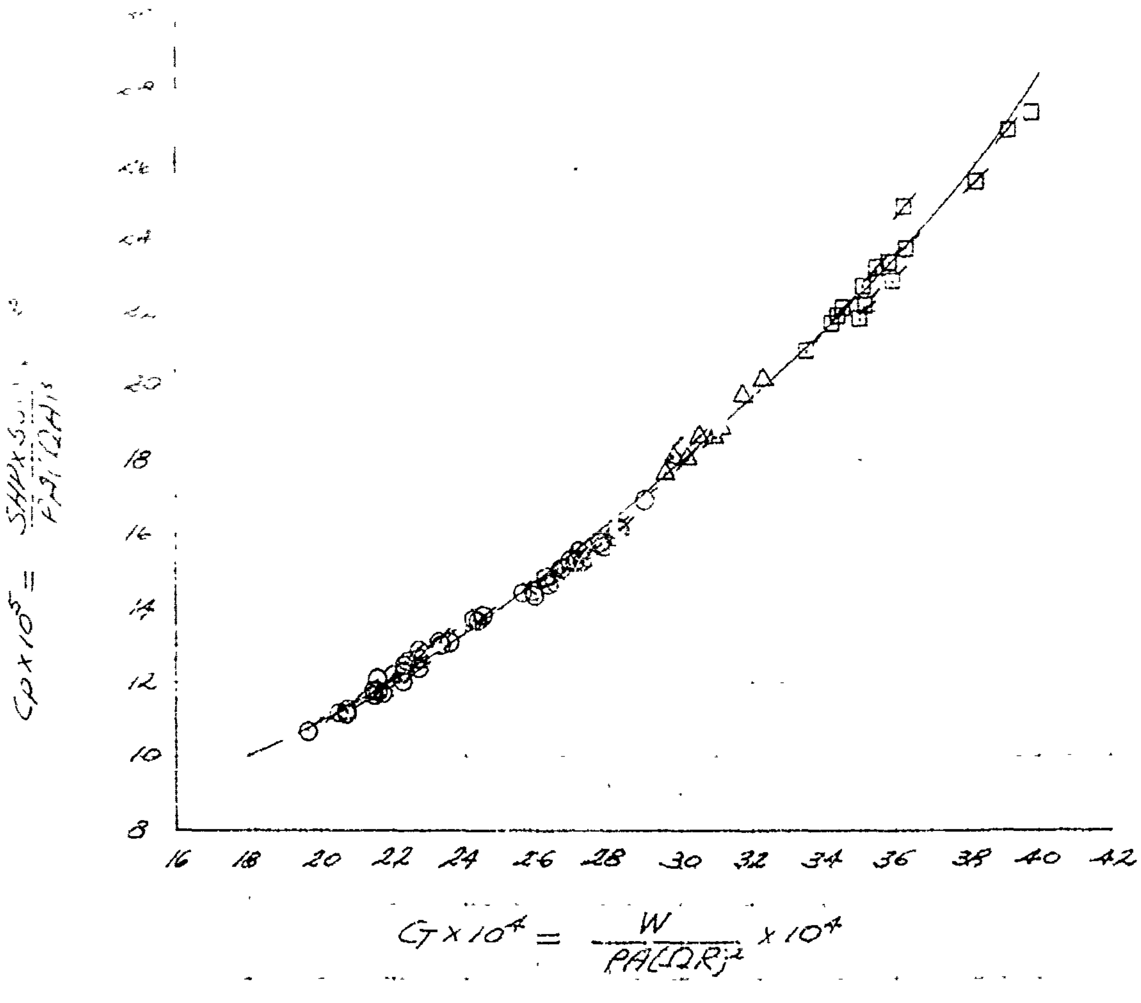
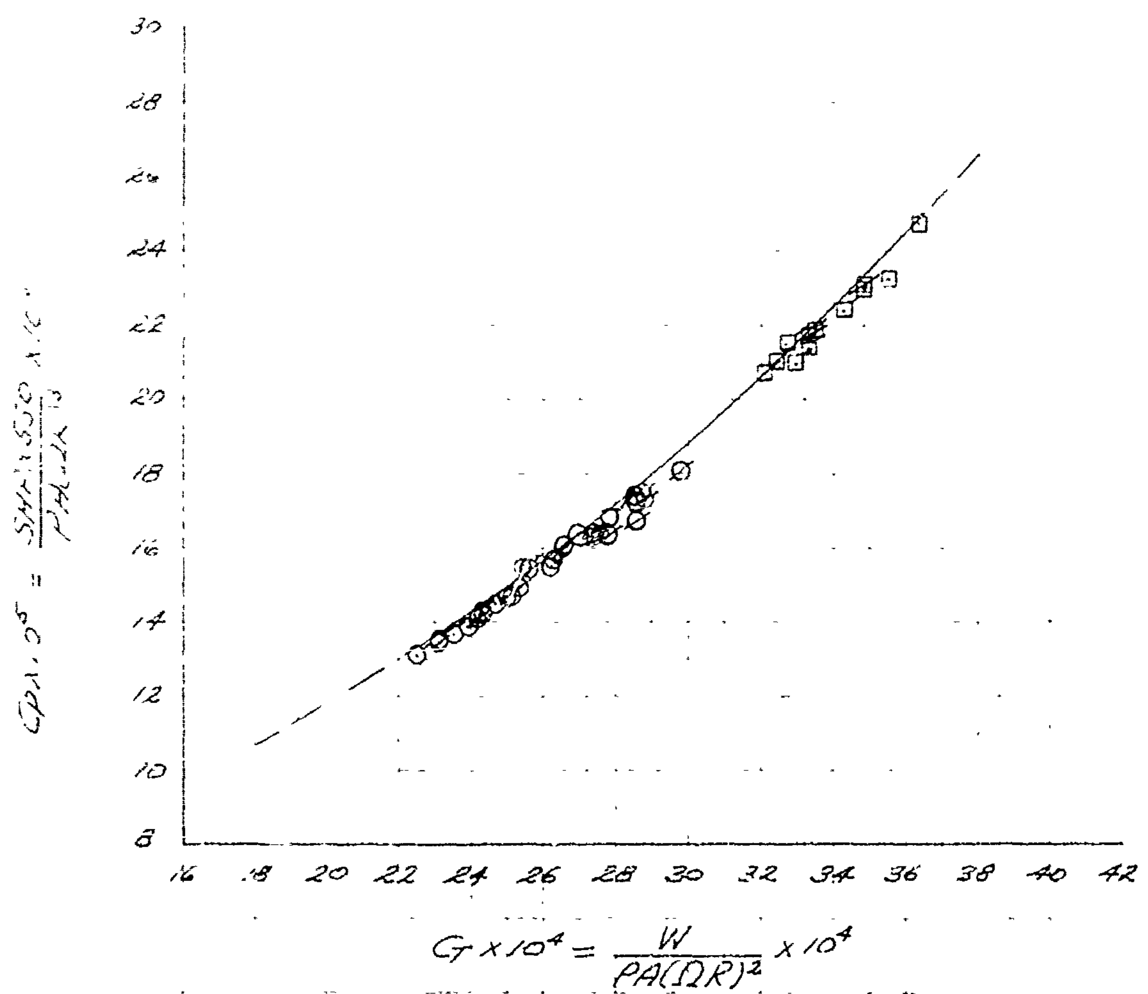


FIGURE No. 27
HOVERING PERFORMANCE
OH-5A USA S/N 62-4201
FREE FLIGHT
SKID HEIGHT = 5 FEET
CLEAN CONFIGURATION

ROTOR SPEED RPM			DENSITY ALT FEET
353	360	368	
○	⊘	○́	380
□	⊠	□́	11143

NOTES
1 VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.77 FEET
2 WIND LESS THAN 2 KNOTS

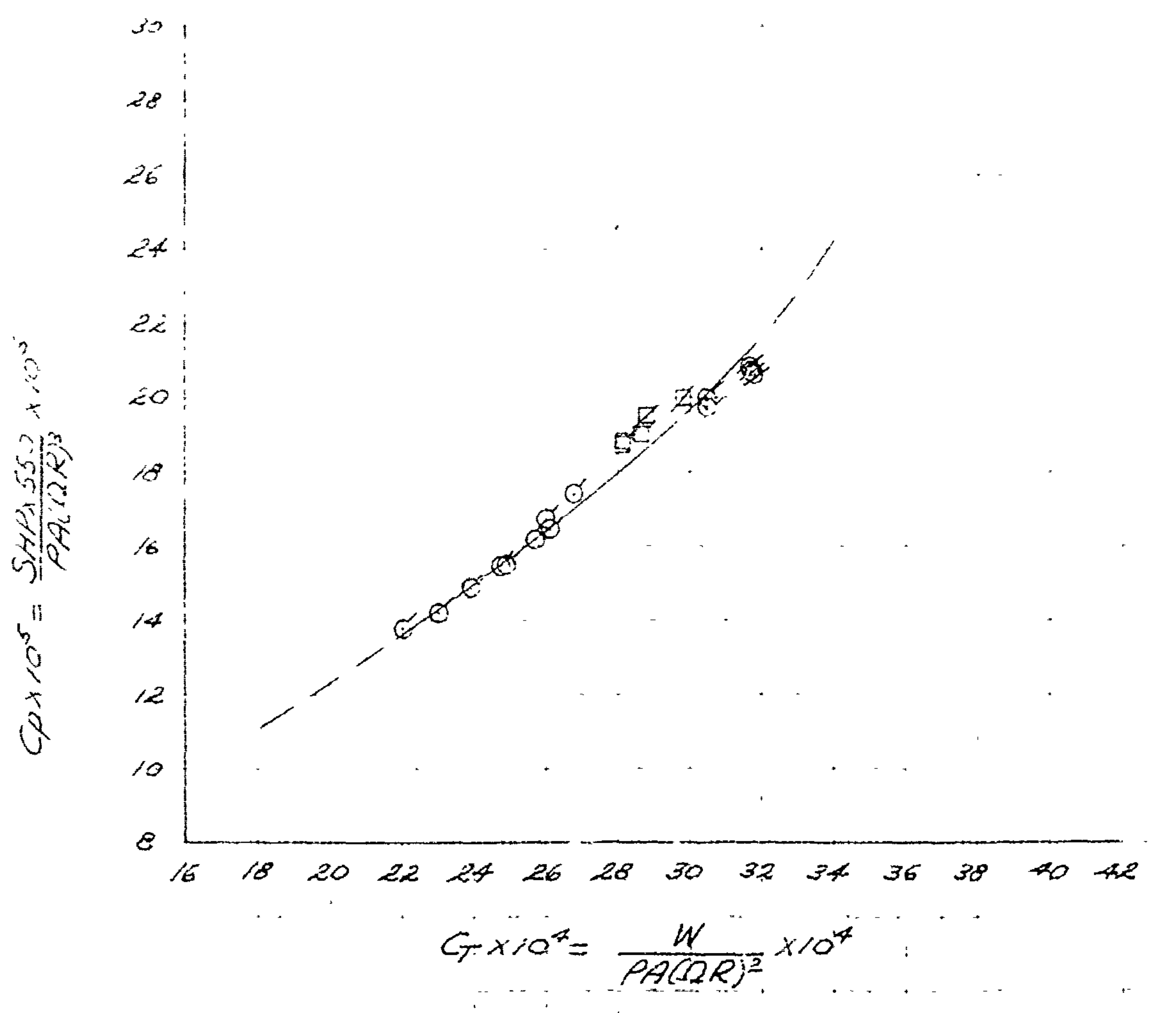


FOR OFFICIAL USE ONLY

FIGURE No 28
HOVERING PERFORMANCE
OH-5A USA S/N 62-4207
FREE FLIGHT
SKID HEIGHT = 10 FEET
CLEAN CONFIGURATION

ROTOR SPEED RPM			DENSITY ALT. FEET
353	360	368	
○	⌀	○	1425
□	□	□	11120

NOTES
1 VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.77 FEET
2 WIND LESS THAN 2 KNOTS



FOR OFFICIAL USE ONLY

FIGURE No 23
HOVERING PERFORMANCE
OH-5A USA S/N 62-4207
FREE FLIGHT
SKID HEIGHT = 50 FT (OGE)
CLEAN CONFIGURATION

ROTOR SPEED ~ RPM ~			DENSITY ALT ~ FEET ~	
355	360	368		
○	○	○	1940	WITH GROUND REFERENCE
△	△	△	4990	
□	□	□	10920	
		□	10620	WITHOUT GROUND REFERENCE
		◇	9800	
		◇	7280	
		◇	5670	

NOTES:
1 VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.77 FEET
2 WIND LESS THAN 2 KNOTS

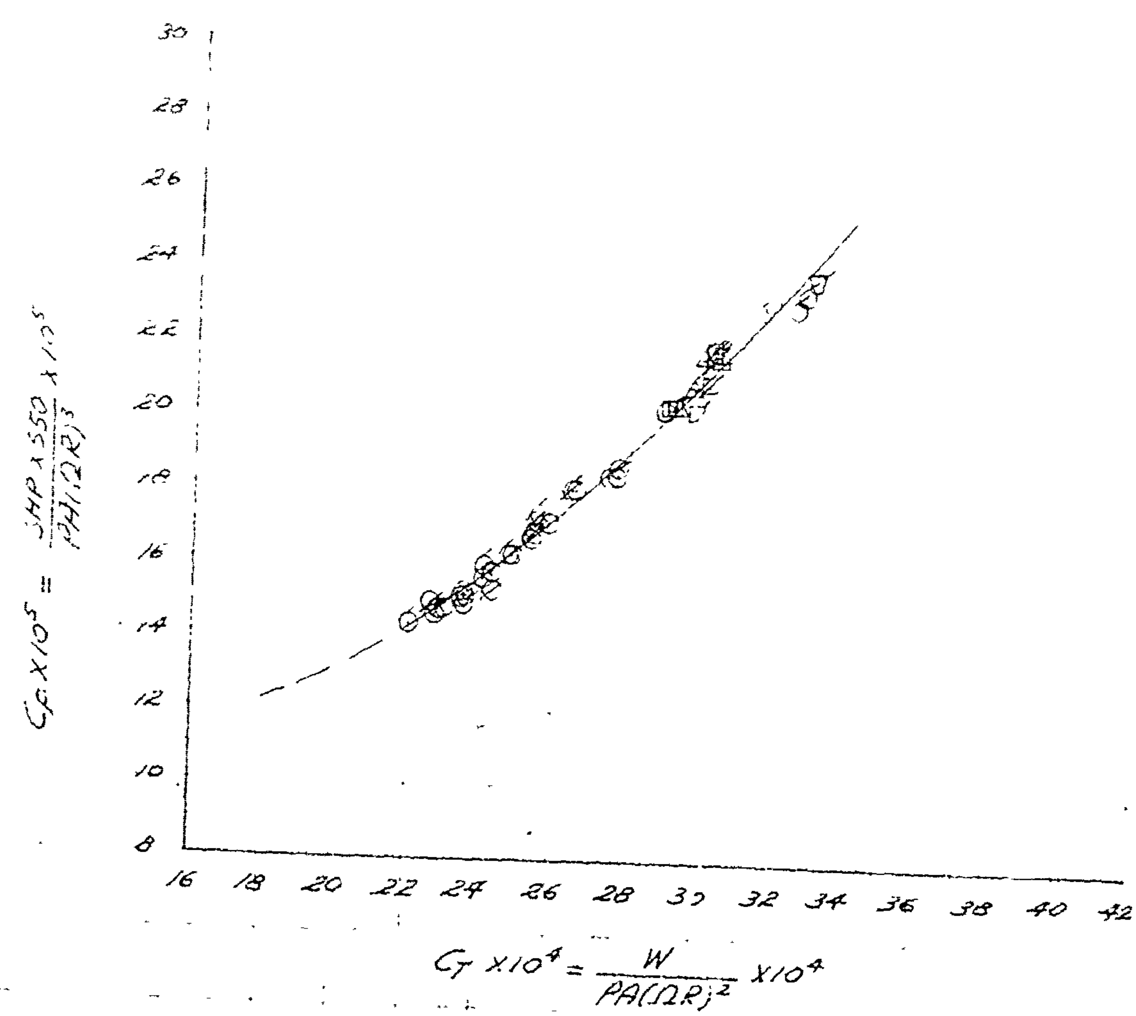
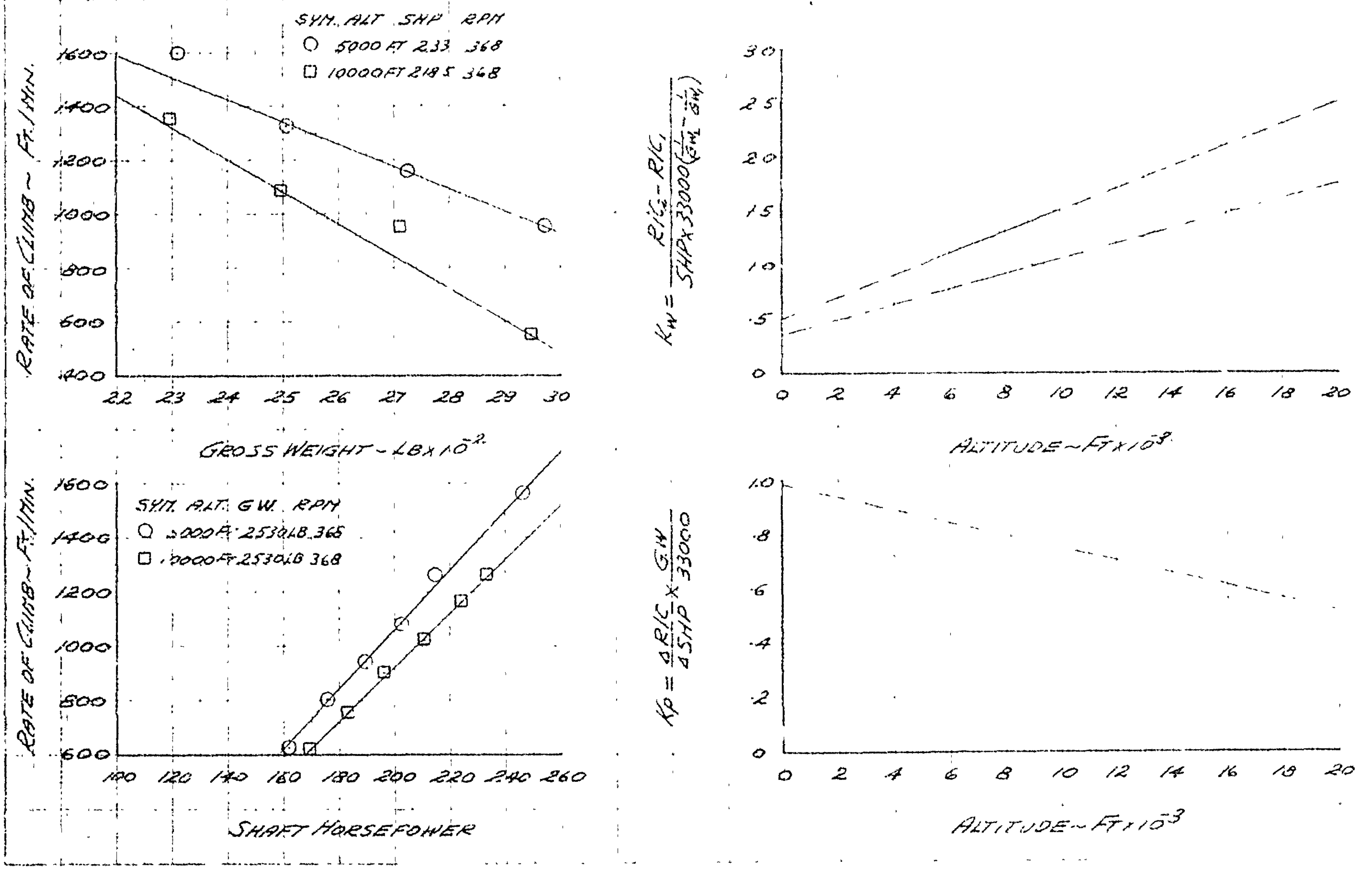


FIGURE NO 30

VARIATION IN RATE OF CLIMB AS A FUNCTION OF GROSS WEIGHT AND SHAFT HORSEPOWER

OH-5A USA S/N 62-4207



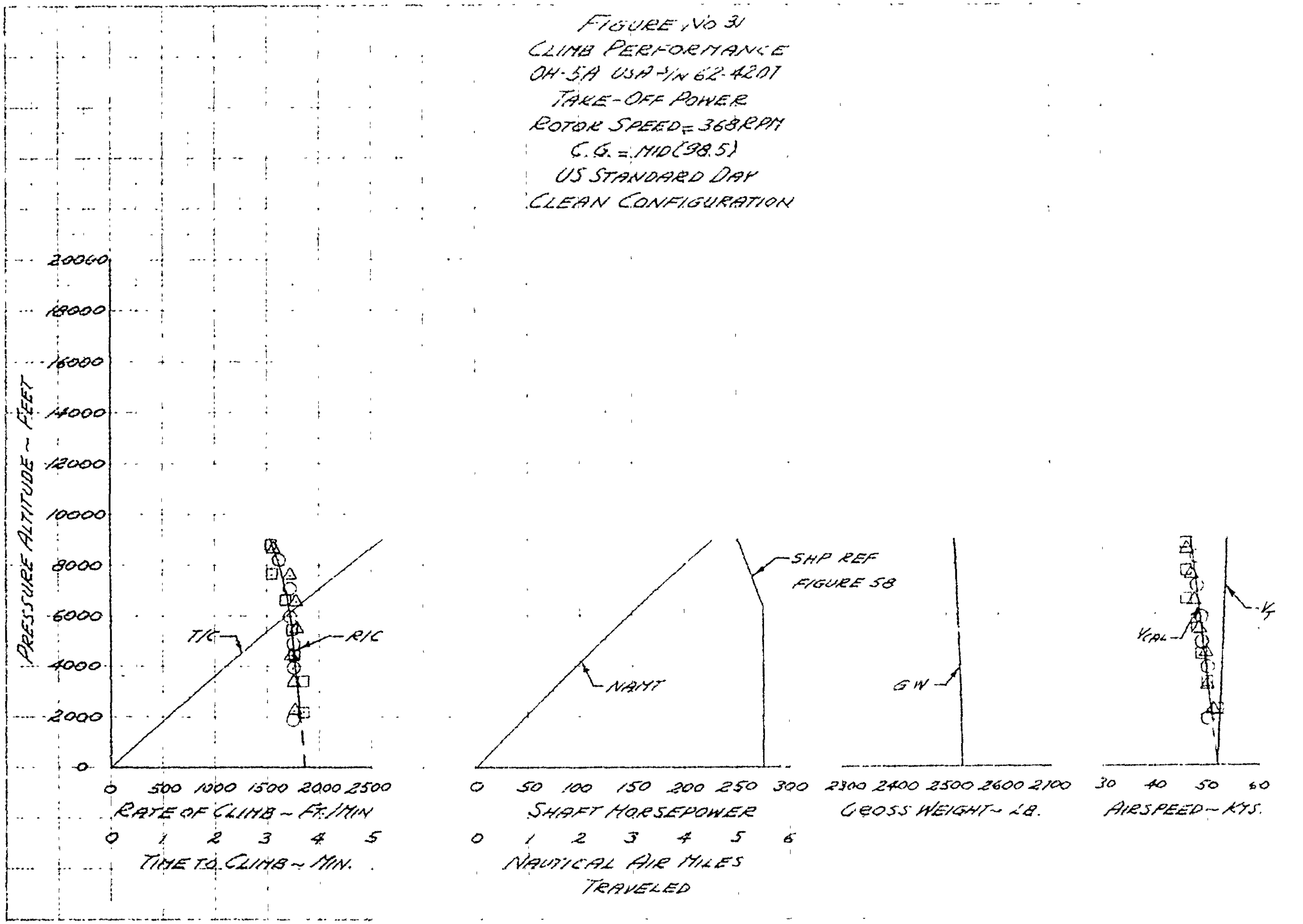


FIGURE NO 31
CLIMB PERFORMANCE
OH-5A USA S/N 62-4201
TAKE-OFF POWER
ROTOR SPEED = 368 RPM
C.G. = MID (98.5)
US STANDARD DAY
CLEAN CONFIGURATION
PRESSURE ALTITUDE ~ FEET
20000
18000
16000
14000
12000
10000
8000
6000
4000
2000
0
T/C
R/C
0 500 1000 1500 2000 2500
RATE OF CLIMB ~ FT./MIN
0 1 2 3 4 5
TIME TO CLIMB ~ MIN.
NAMT
SHP REF FIGURE 58
0 50 100 150 200 250 300
SHAFT HORSEPOWER
0 1 2 3 4 5 6
NAUTICAL AIR MILES TRAVELED
GW
2300 2400 2500 2600 2700
GROSS WEIGHT ~ LB.
VCAL
VT
30 40 50 60
AIRSPEED ~ KTS.

FIGURE No. 32
CLIMB PERFORMANCE
OH-5A USA S/N 62-4207
MAXIMUM CONTINUOUS POWER
ROTOR SPEED = 368 RPM
C.G. = MID (98.5)
US STANDARD DAY
CLEAN CONFIGURATION

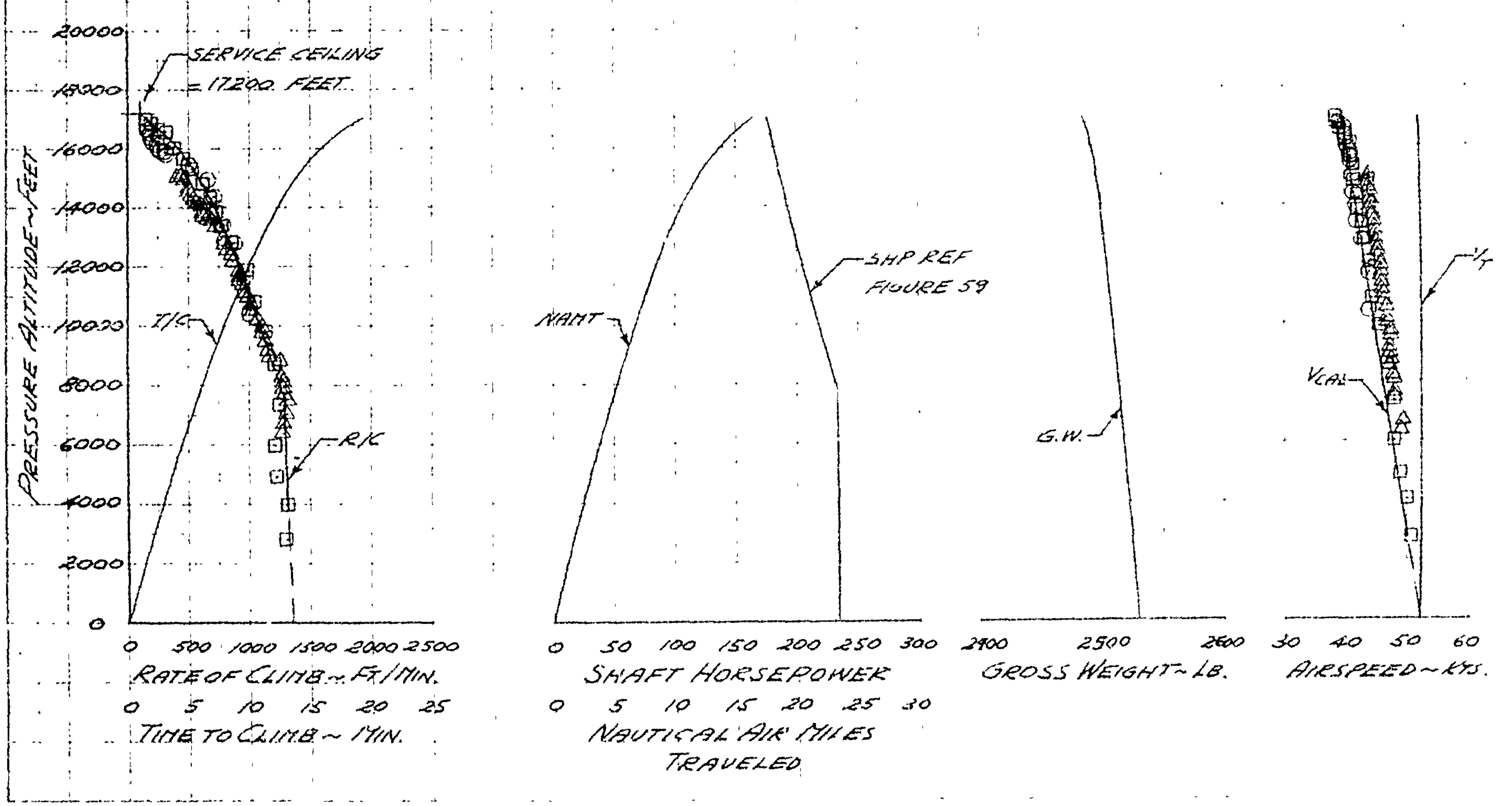


FIGURE NO. 33
CLIMB PERFORMANCE
OH-5A USA S/N 62-4207
MAXIMUM CONTINUOUS POWER
ROTOR SPEED = 368 RPM
C.G. = MID (98.5)
US STANDARD DAY
CLEAN CONFIGURATION

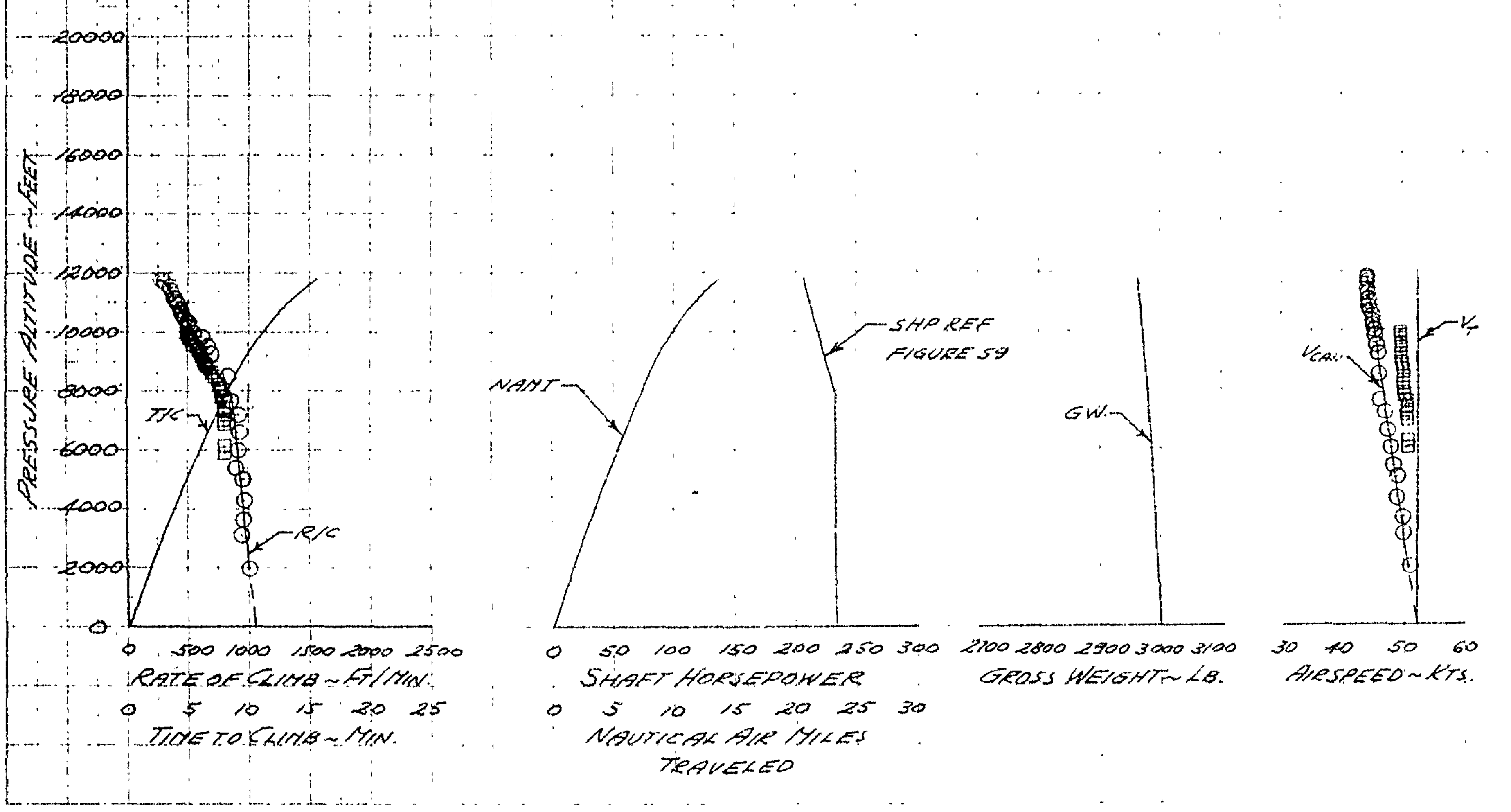


FIGURE NO. 34
LEVEL FLIGHT SUMMARY
OH-5A USA S/N 62-4207

SYM	RPM	CONFIGURATION
○	368	CLEAN
□	353	CLEAN
△	368	FWD C.G
▽	368	AFT C G
⬡	368	XM-7
◇	368	XM-8
▽	368	ANT OFF

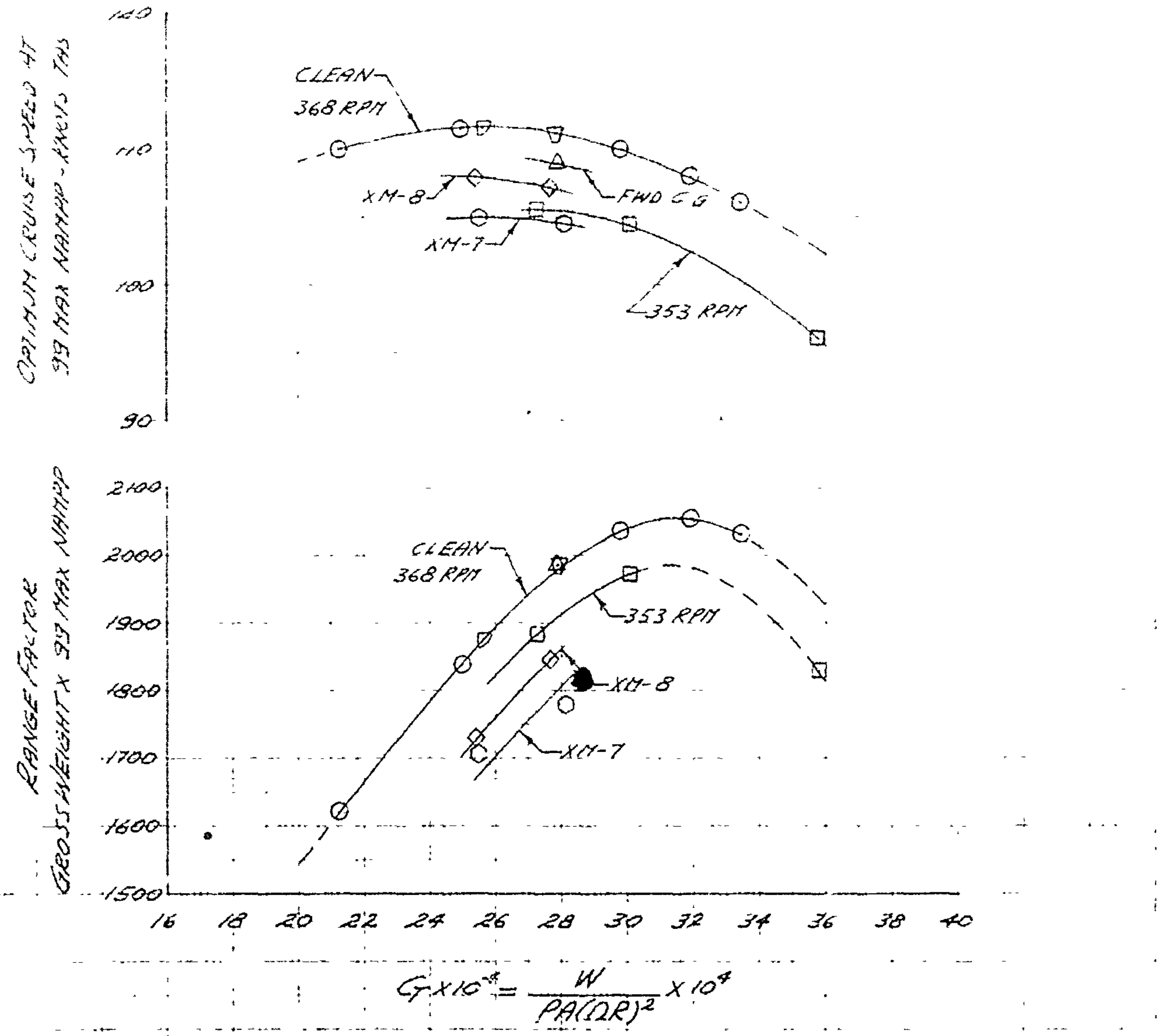
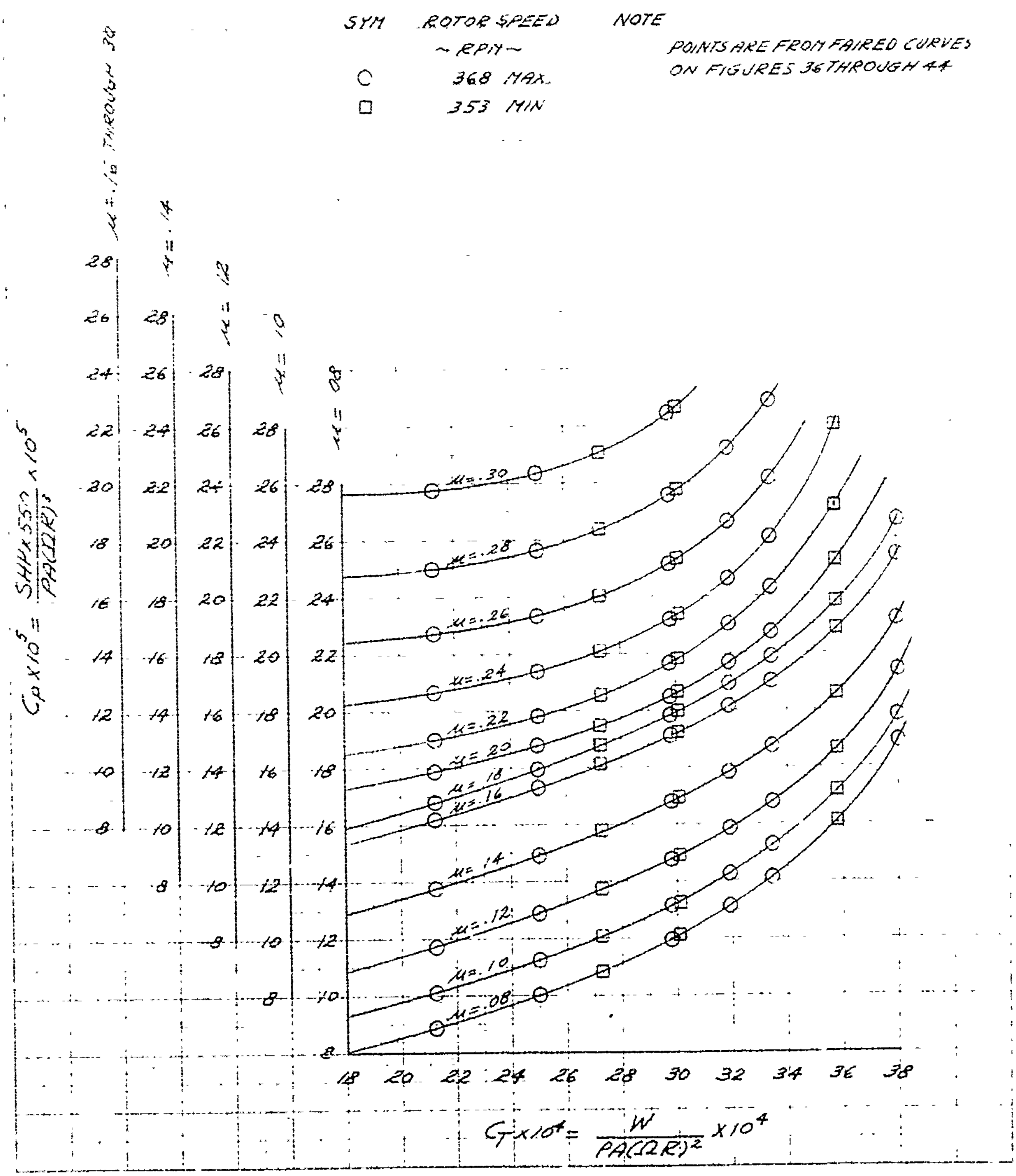


FIGURE No. 35
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
C.G. = STA. 98.5 (MID)
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE NO. 36
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2160 LB.
DENSITY ALTITUDE = 900 FT.
ROTOR SPEED = 368 RPM
C.G. = MID (98.7)
C_T = .002125
CLEAN CONFIGURATION

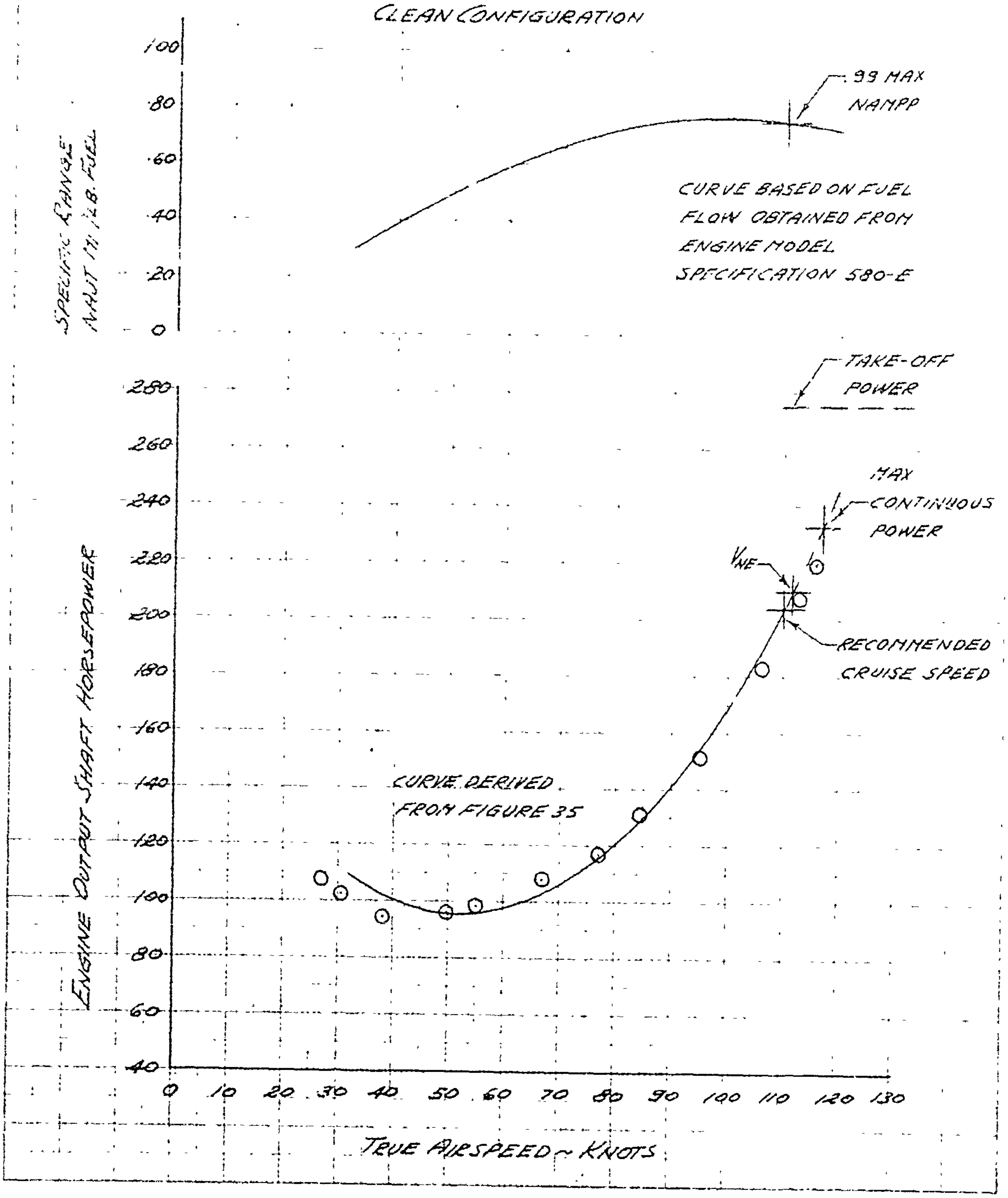


FIGURE NO. 37

LEVEL FLIGHT PERFORMANCE

OH-5A USA S/N 62-4207

GROSS WEIGHT = 2520 LB.

DENSITY ALTITUDE = 950 FT

ROTOR SPEED = 368 RPM

C.G. = MID (98.7)

C_T = .002499

CLEAN CONFIGURATION

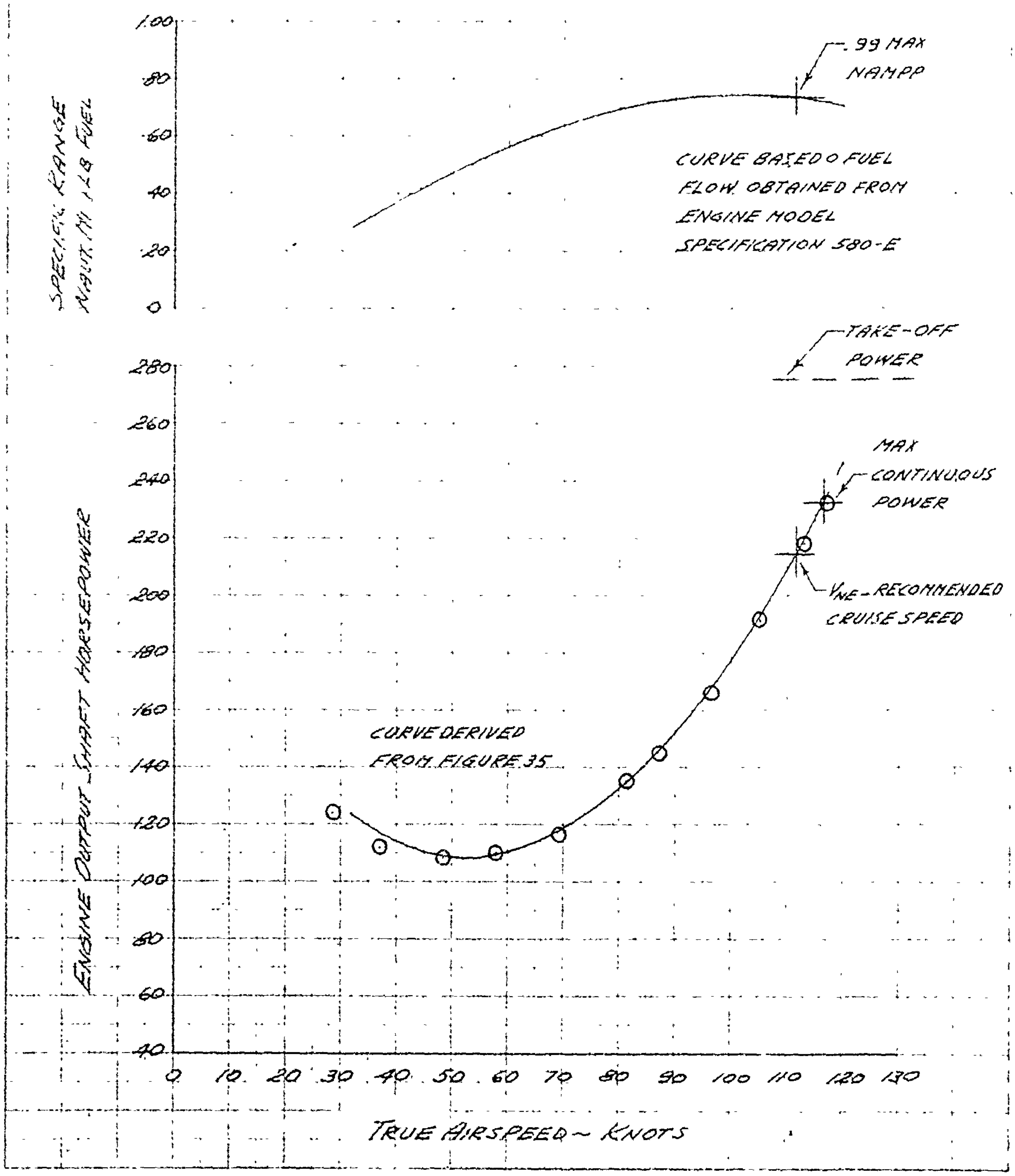


FIGURE NO. 38
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2510 LB
DENSITY ALTITUDE = 1190 FT
ROTOR SPEED = 353 RPM
C.G = MID (98.5)
C_T = .002727
CLEAN CONFIGURATION

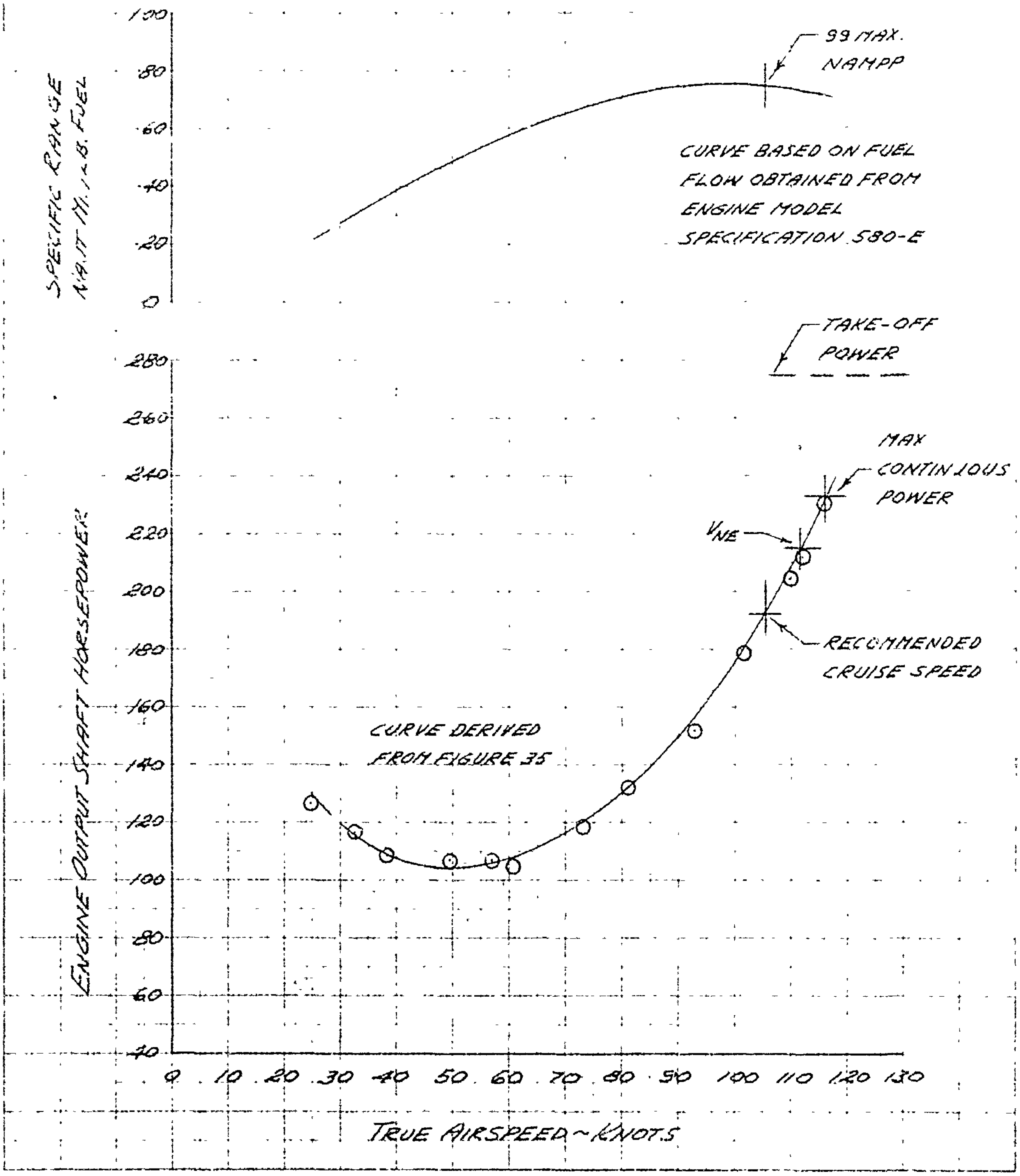


FIGURE No. 39
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2940 LB.
DENSITY ALTITUDE = 1750 FT.
ROTOR SPEED = 368 RPM
C.G. = MID (98.75)
C_T = .002980
CLEAN CONFIGURATION

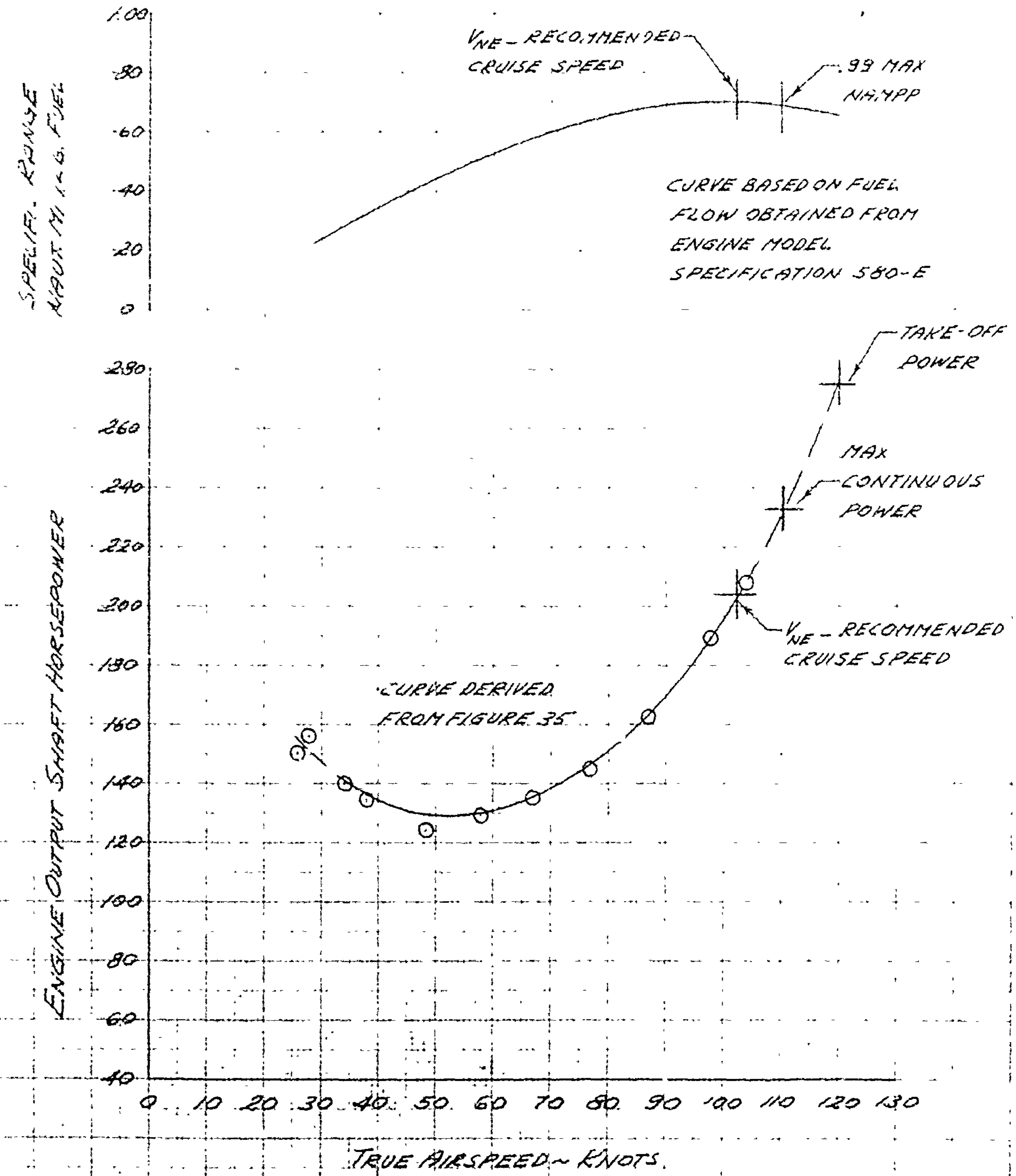


FIGURE NO. 40
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2480 LB
DENSITY ALTITUDE = 5100 FT.
ROTOR SPEED = 353 RPM
C.G. = MID (98.6)
C_T = .003007
CLEAN CONFIGURATION

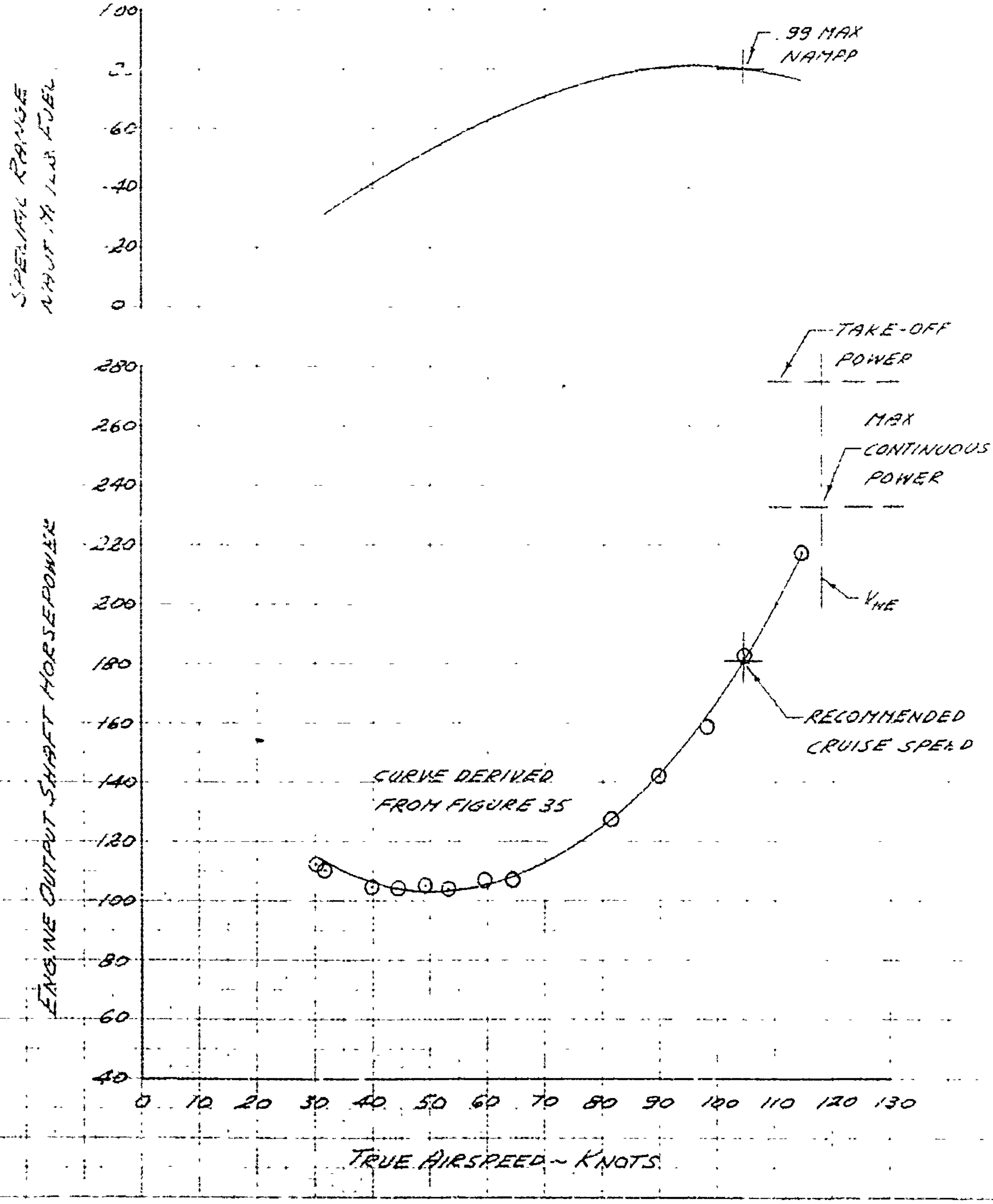


FIGURE NO. 41
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2490 LB
DENSITY ALTITUDE = 9320 FT
ROTOR SPEED = 368 RPM
C.G. = MID (98.6)
C_T = .003193
CLEAN CONFIGURATION

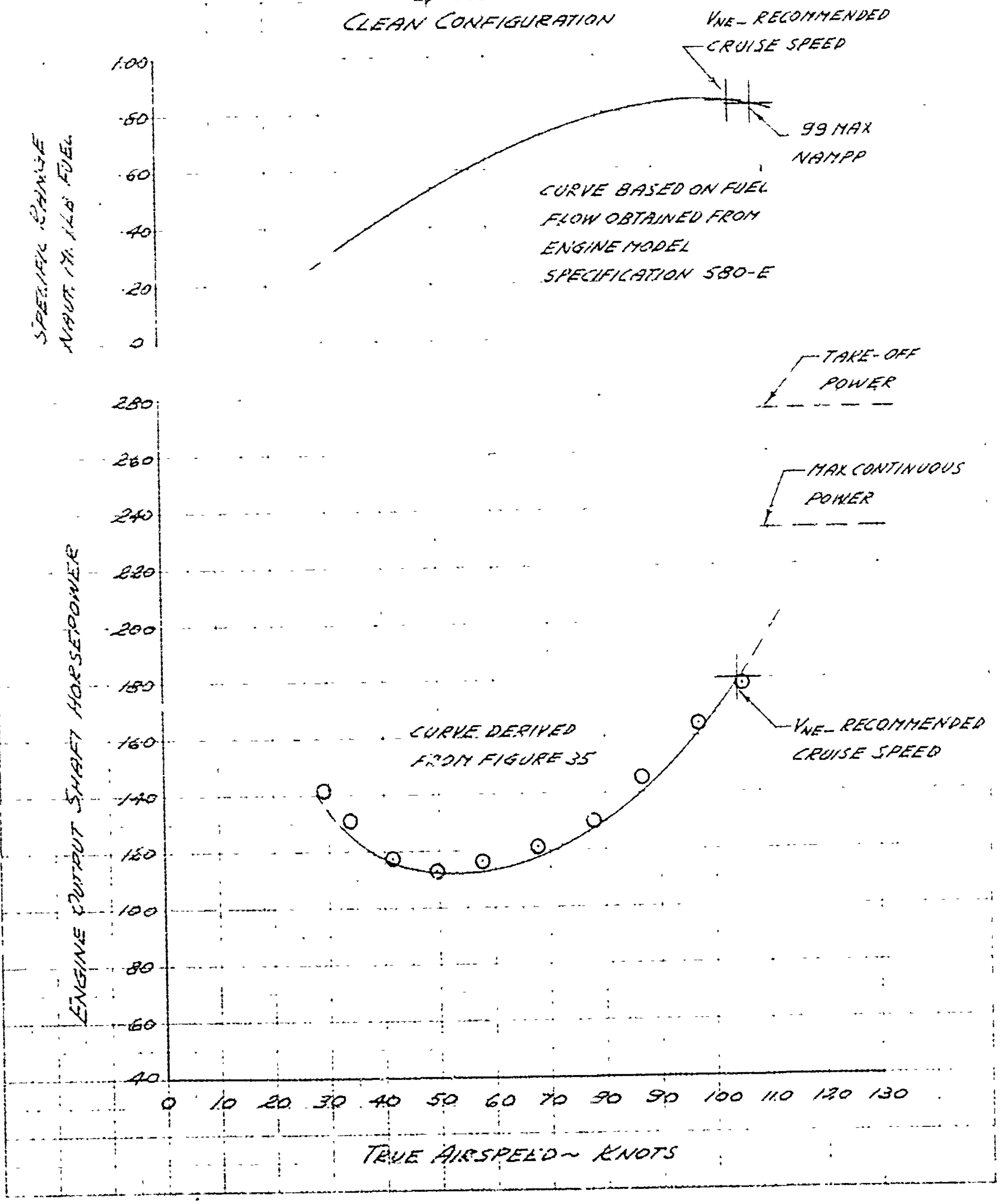


FIGURE NO 42
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2900 LB.
DENSITY ALTITUDE = 5860 FT
ROTOR SPEED = 368 RPM
C G = MID (98.7)
C_T = .003343
CLEAN CONFIGURATION

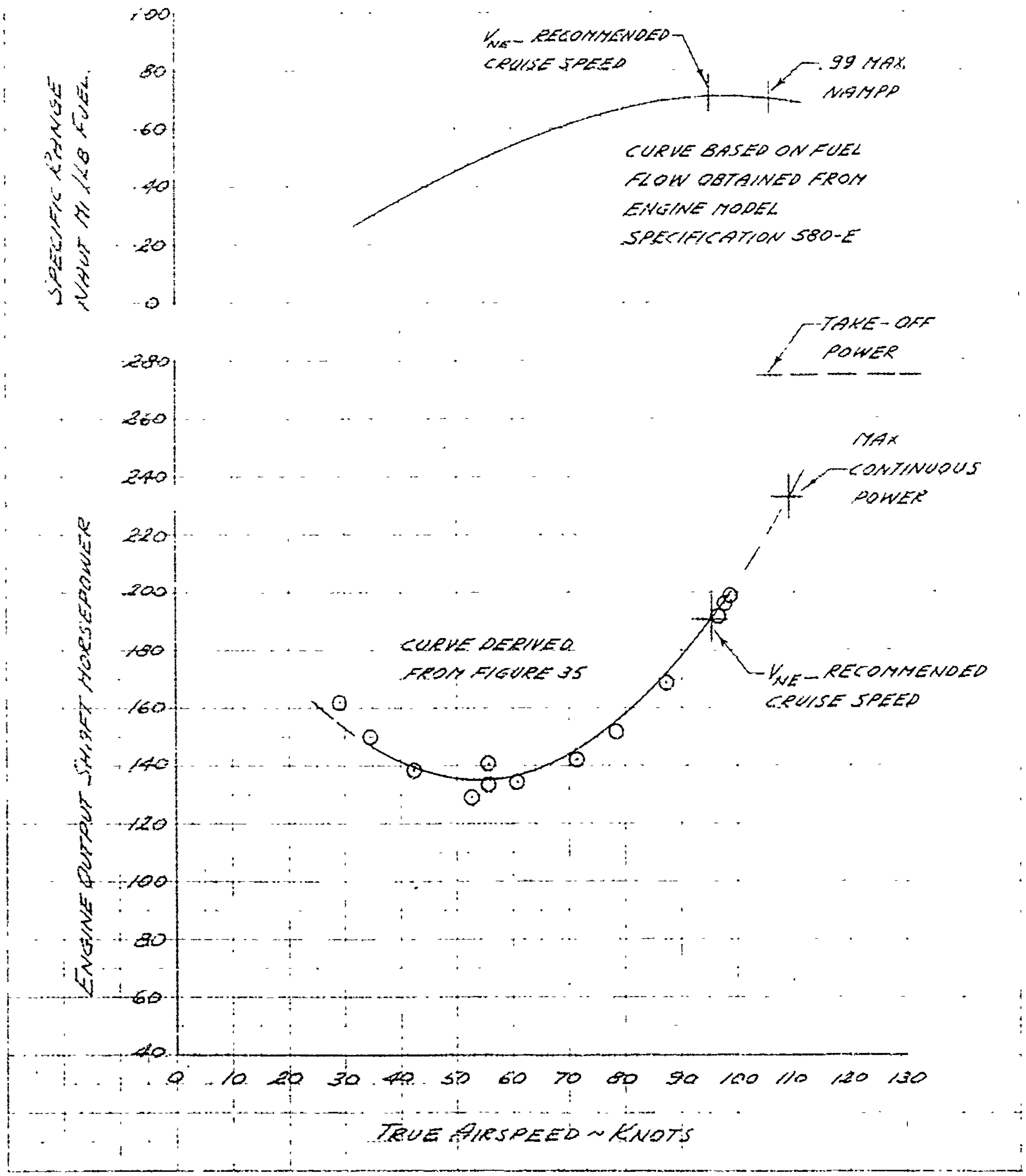
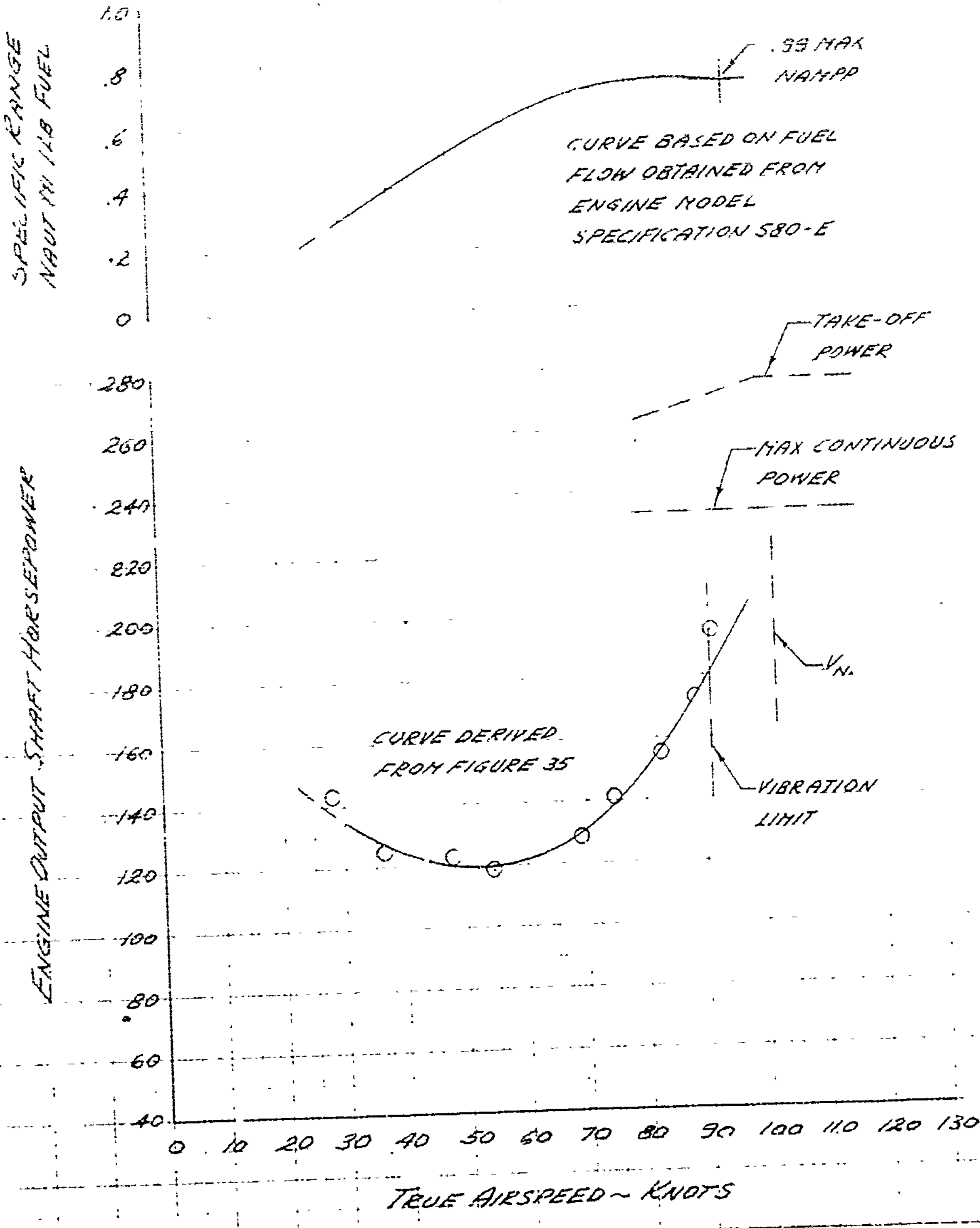


FIGURE NO 43
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2540 LB.
DENSITY ALTITUDE = 9893 FT
ROTOR SPEED = 353 RPM
C.G. = 1110 (98.9)
C_T = .003579
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE No. 44
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2500 LB.
DENSITY ALTITUDE = 14675 FT.
ROTOR SPEED = 368 RPM
C.G. = MID (98.6)
C_T = .003796
CLEAN CONFIGURATION

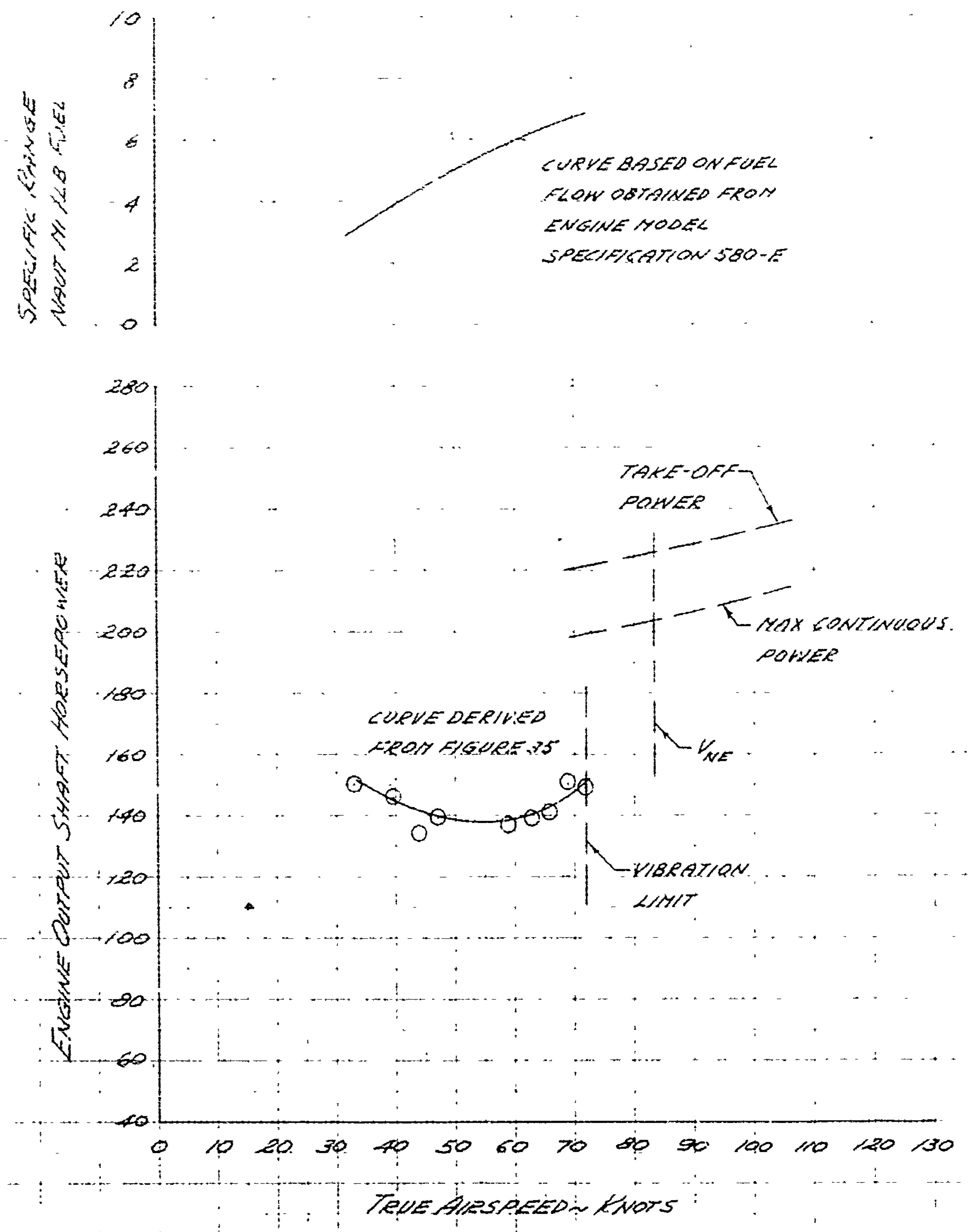


FIGURE No. 45
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2470 LB.
DENSITY ALTITUDE = 2395 FT.
ROTOR SPEED = 368 RPM
C.G. = MID (98.3)
C_T = .002561
ANTENNA OFF

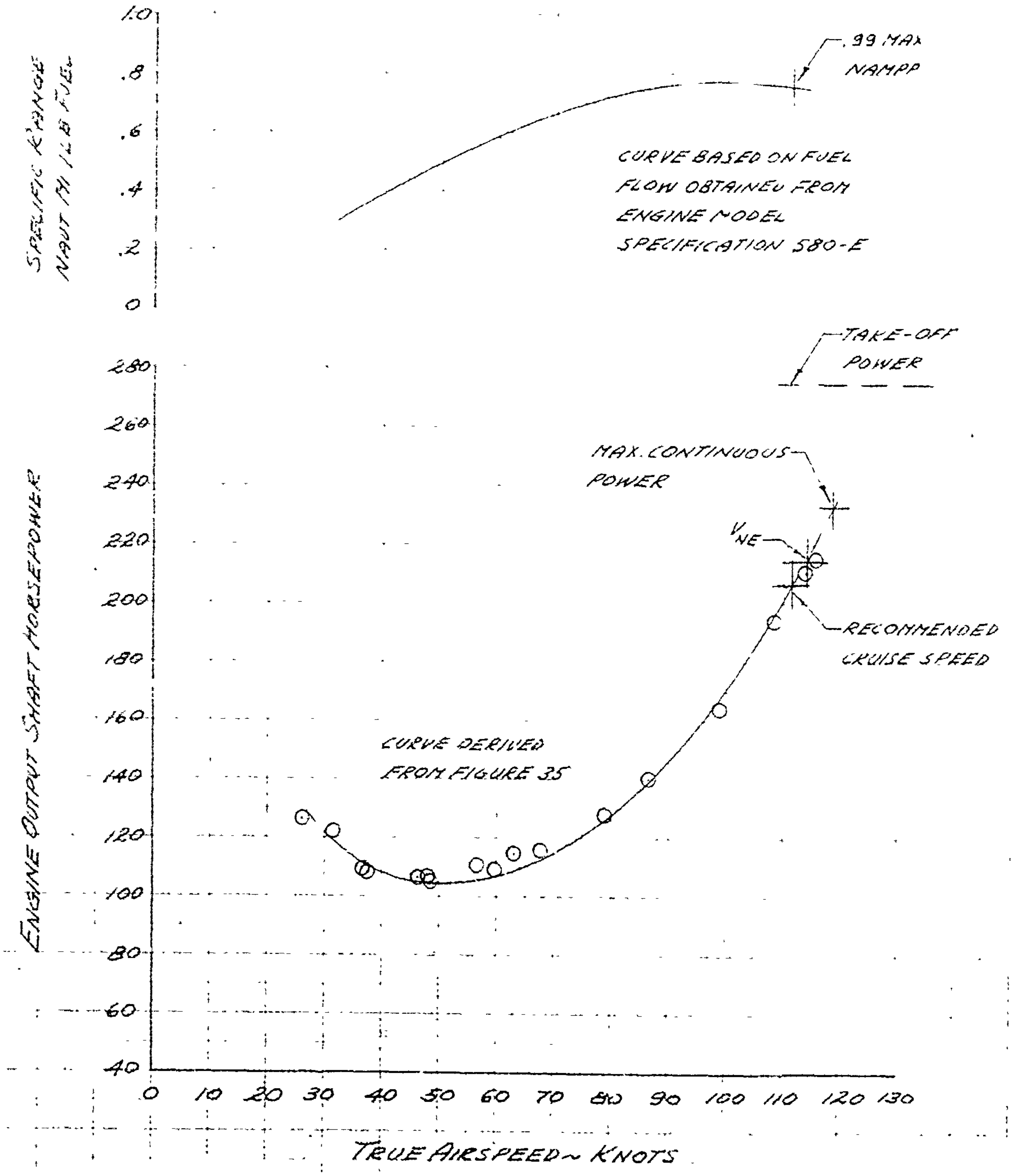


FIGURE NO 46
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2480 LB
DENSITY ALTITUDE = 9860 FT
ROTOR SPEED = 368 RPM
C.G. = MID (98.6)
C_T = .003230
ANTENNA OFF

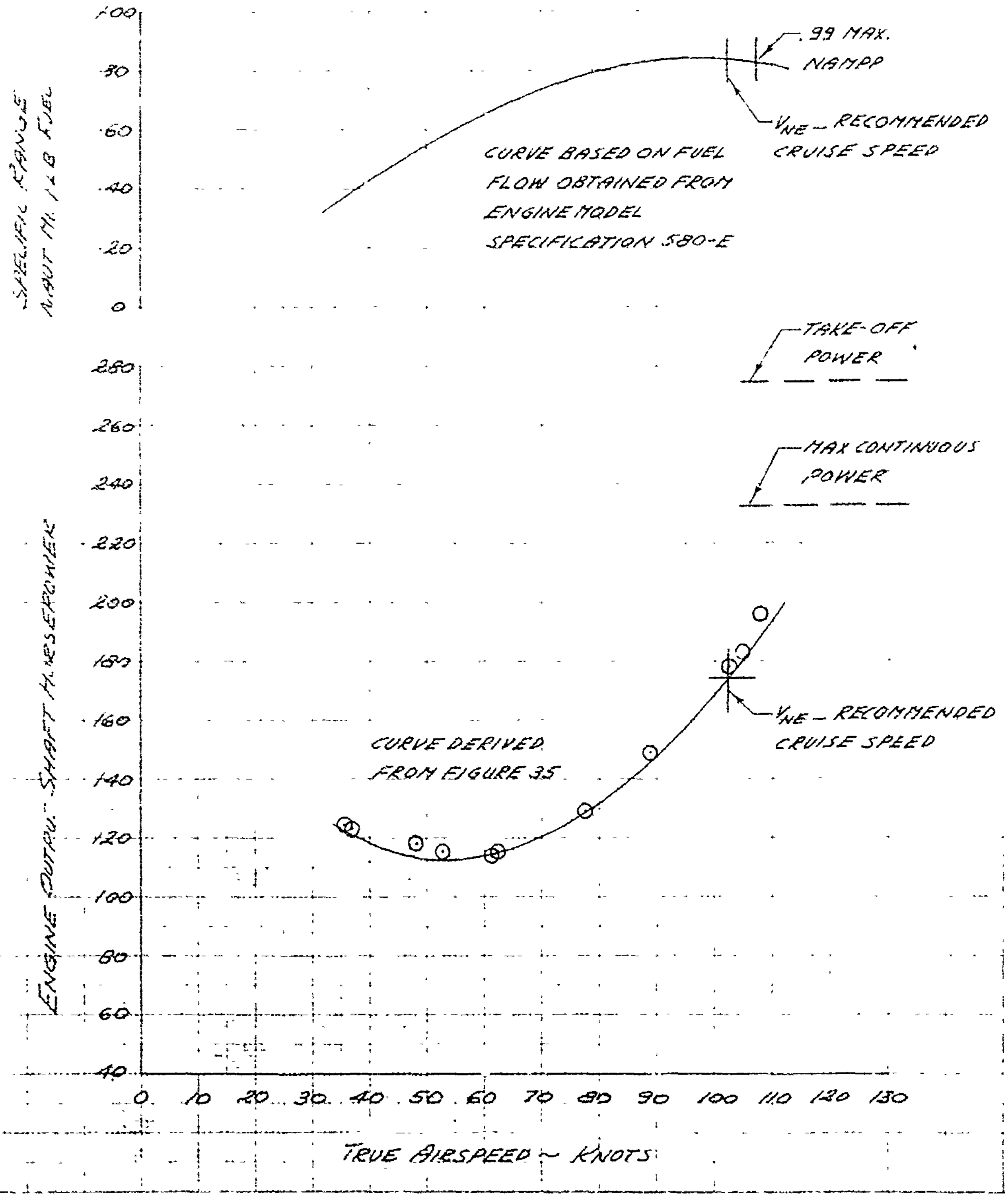


FIGURE No 47
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2480 LB.
DENSITY ALTITUDE = 5200 FT
ROTOR SPEED = 368 RPM
C.G. = FWD (95.8)
C_T = .002788
CLEAN CONFIGURATION

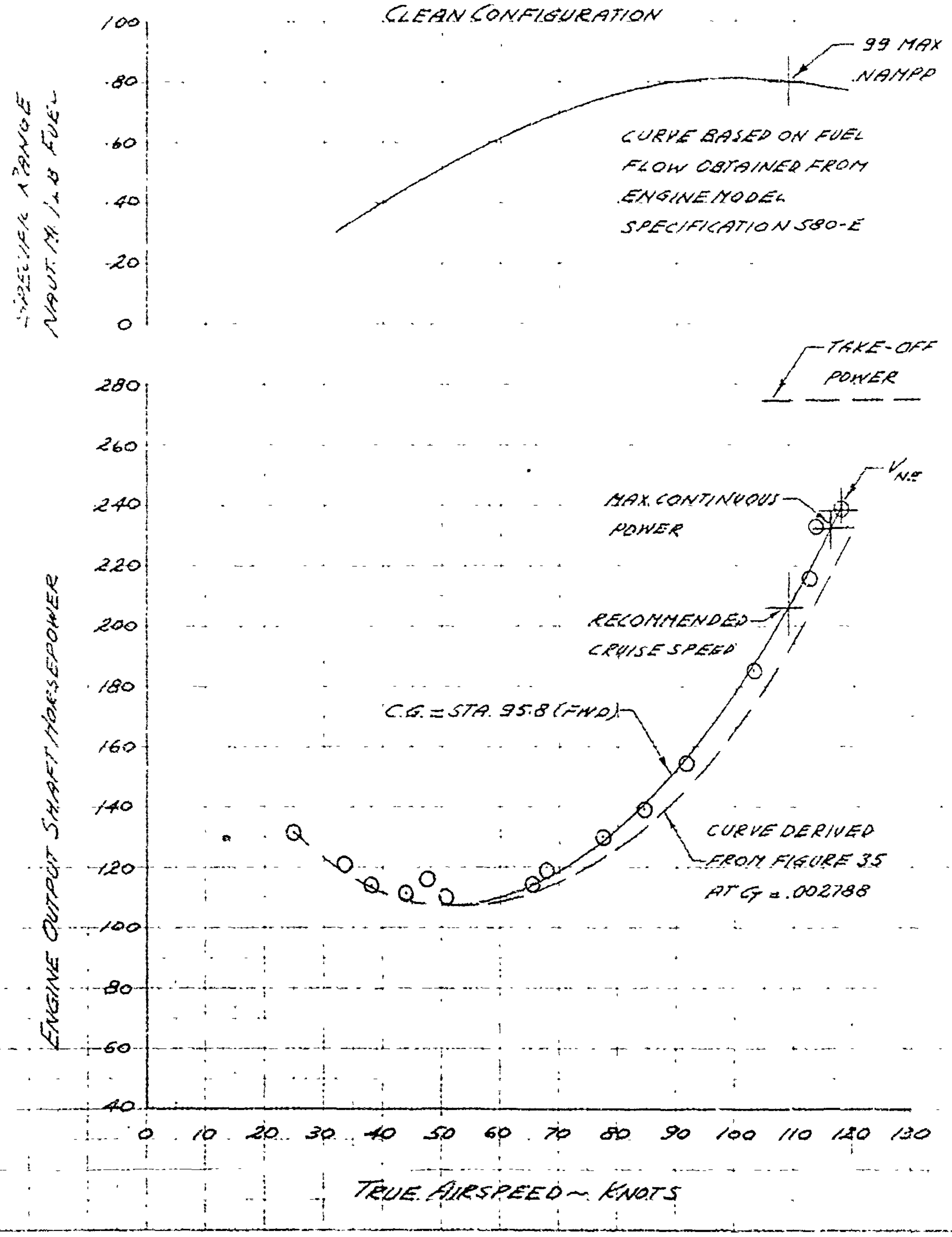
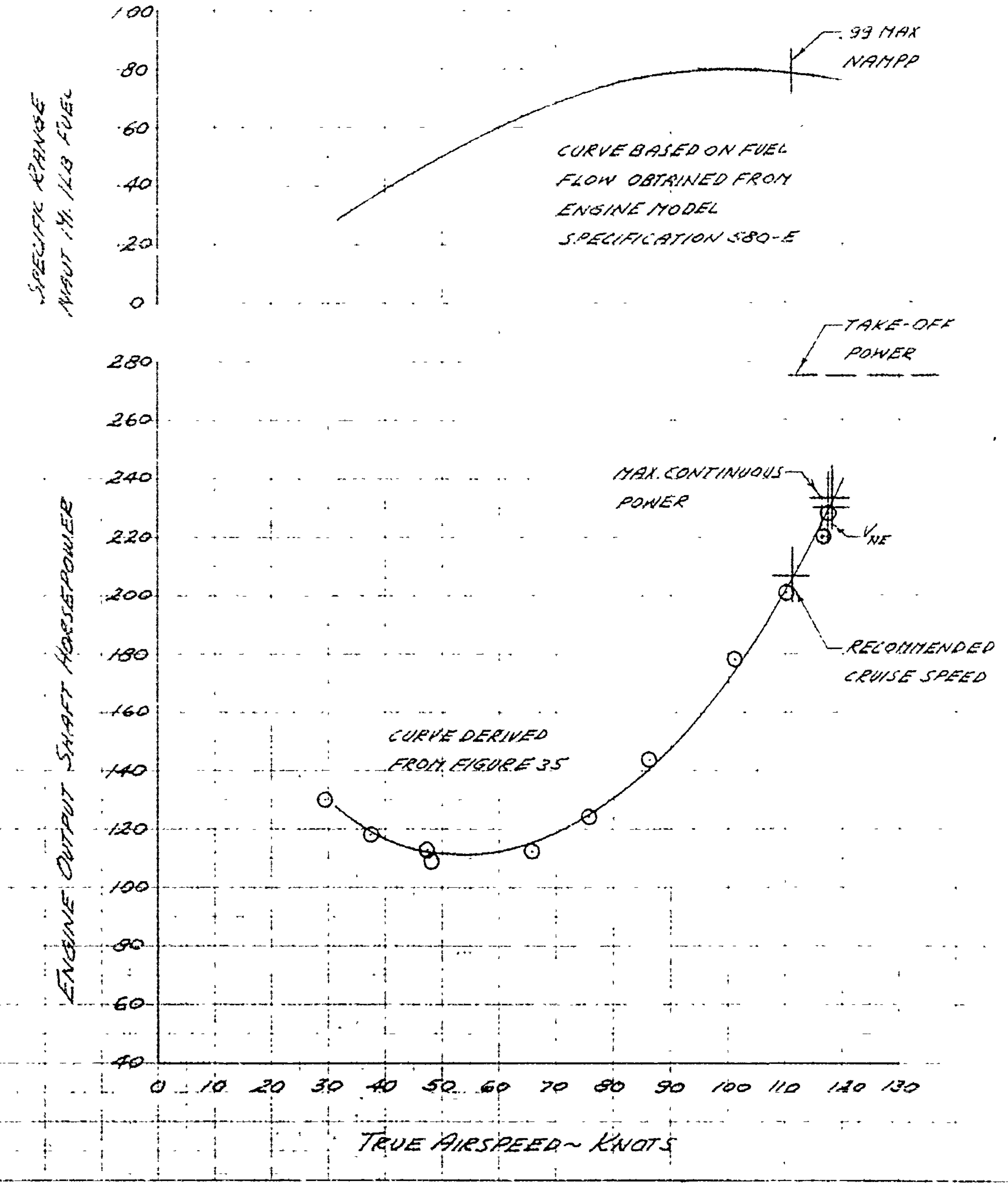


FIGURE NO. 48
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2540 LB.
DENSITY ALTITUDE = 4340 FT
ROTOR SPEED = 368 RPM
C.G. = AFT (100.7)
C_T = .002789
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

FIGURE NO. 49
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2550 LB.
DENSITY ALTITUDE = 1230 FT.
ROTOR SPEED = 368 RPM
C.G. = MID (98.6)
C_T = .002549
XM-7 ARMAMENT SYSTEM
INSTALLED IN STOWED POSITION

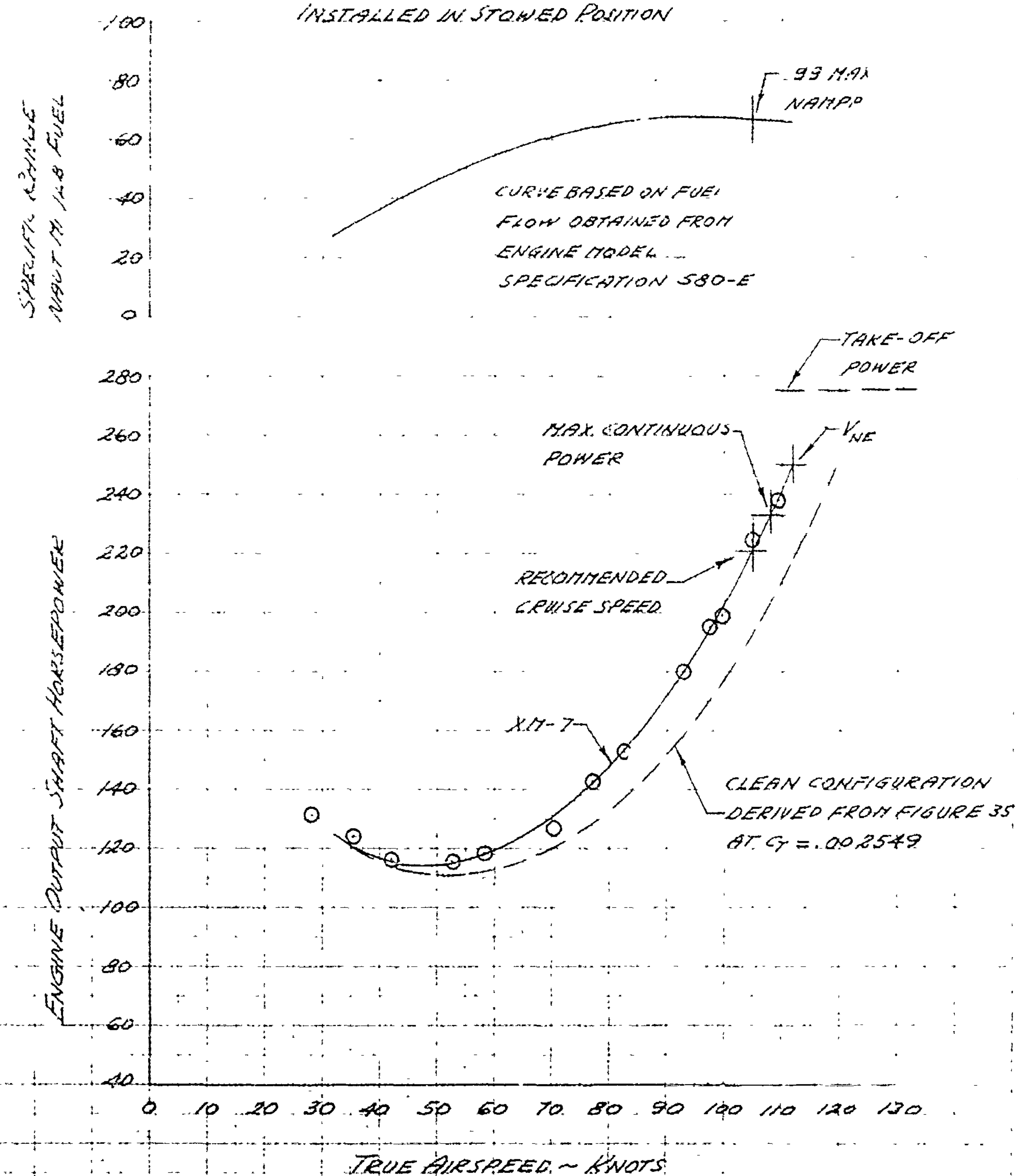
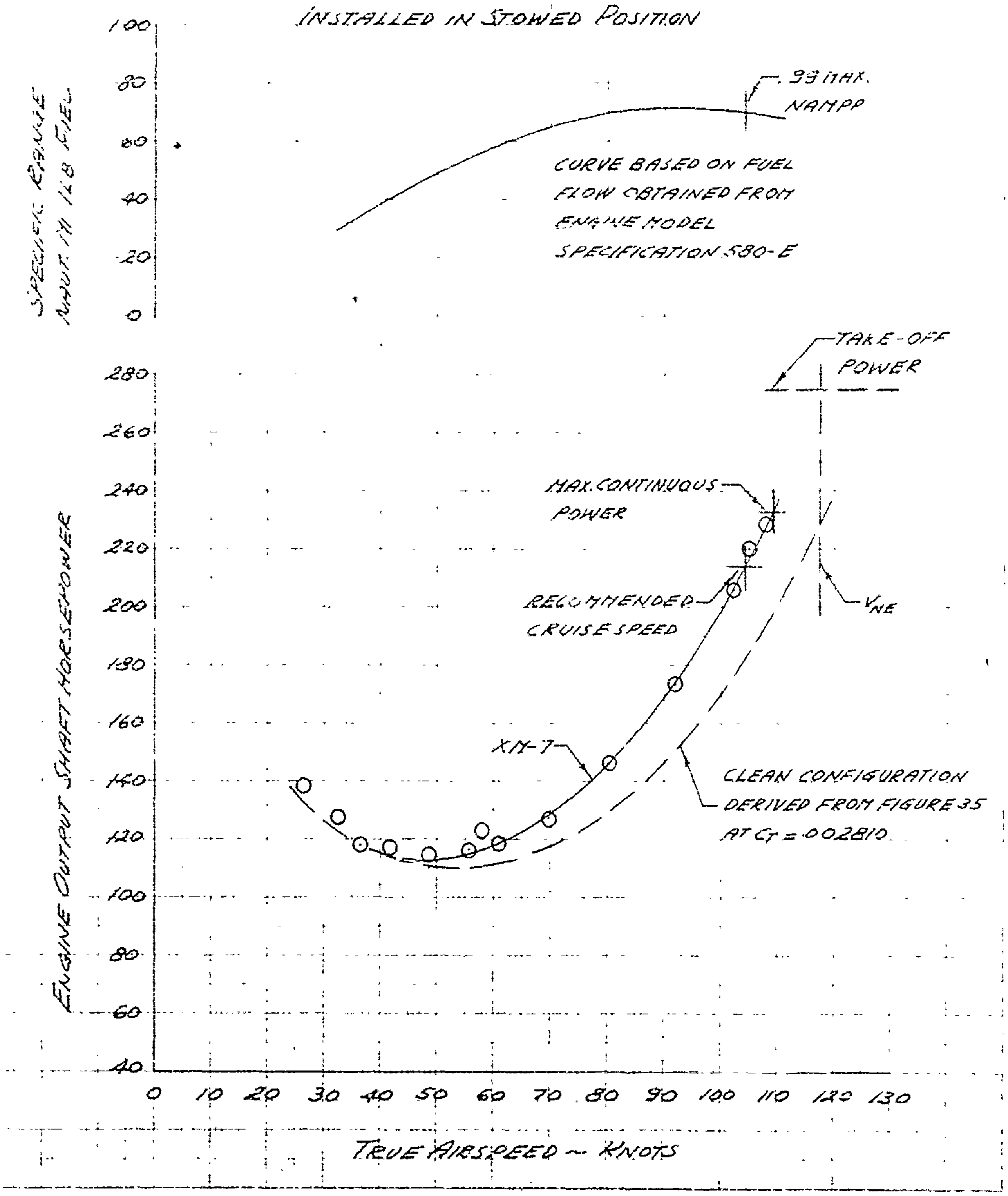


FIGURE NO. 50
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2540 LB.
DENSITY ALTITUDE = 4530 FT.
ROTOR SPEED = 368 RPM
C.G. = MID (98.5)
C_T = .002810
XM-7 ARMAMENT SYSTEM
INSTALLED IN STOWED POSITION



FOR [illegible] ONLY

FIGURE NO. 51
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2540 LB.
DENSITY ALTITUDE = 1620 FT
ROTOR SPEED = 368 RPM.
C.G. = MID. (98.6)
C_T = .002582
XM-8 ARMAMENT SYSTEM
INSTALLED IN STOWED POSITION

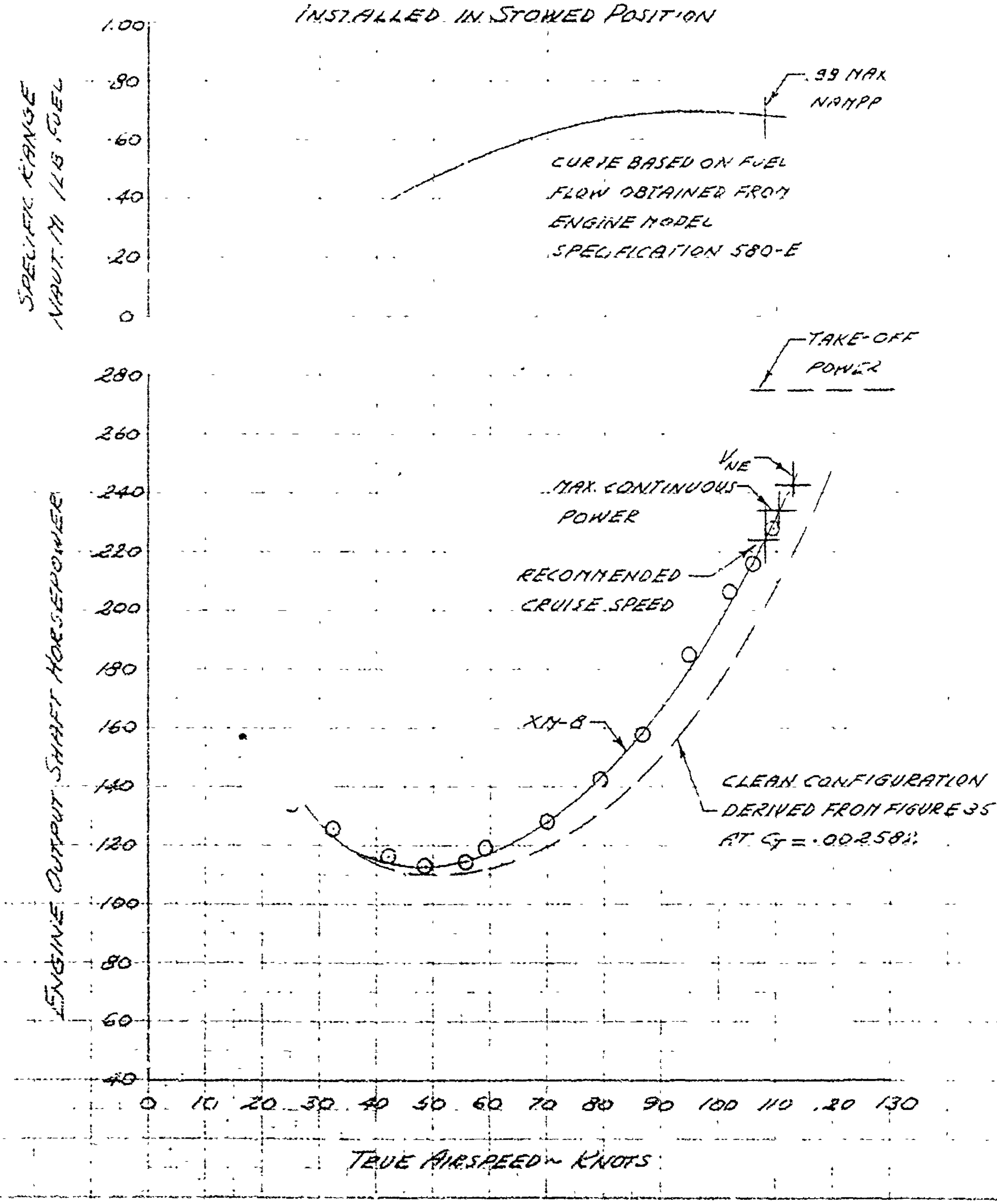


FIGURE No 52
LEVEL FLIGHT PERFORMANCE
OH-5A USA S/N 62-4207
GROSS WEIGHT = 2530 LB.
DENSITY ALTITUDE = 4270 FT.
ROTOR SPEED = 368 RPM
C.G. = MID (98.6)
C_T = .002766
XM-8 ARMAMENT SYSTEM
INSTALLED IN STOWED POSITION

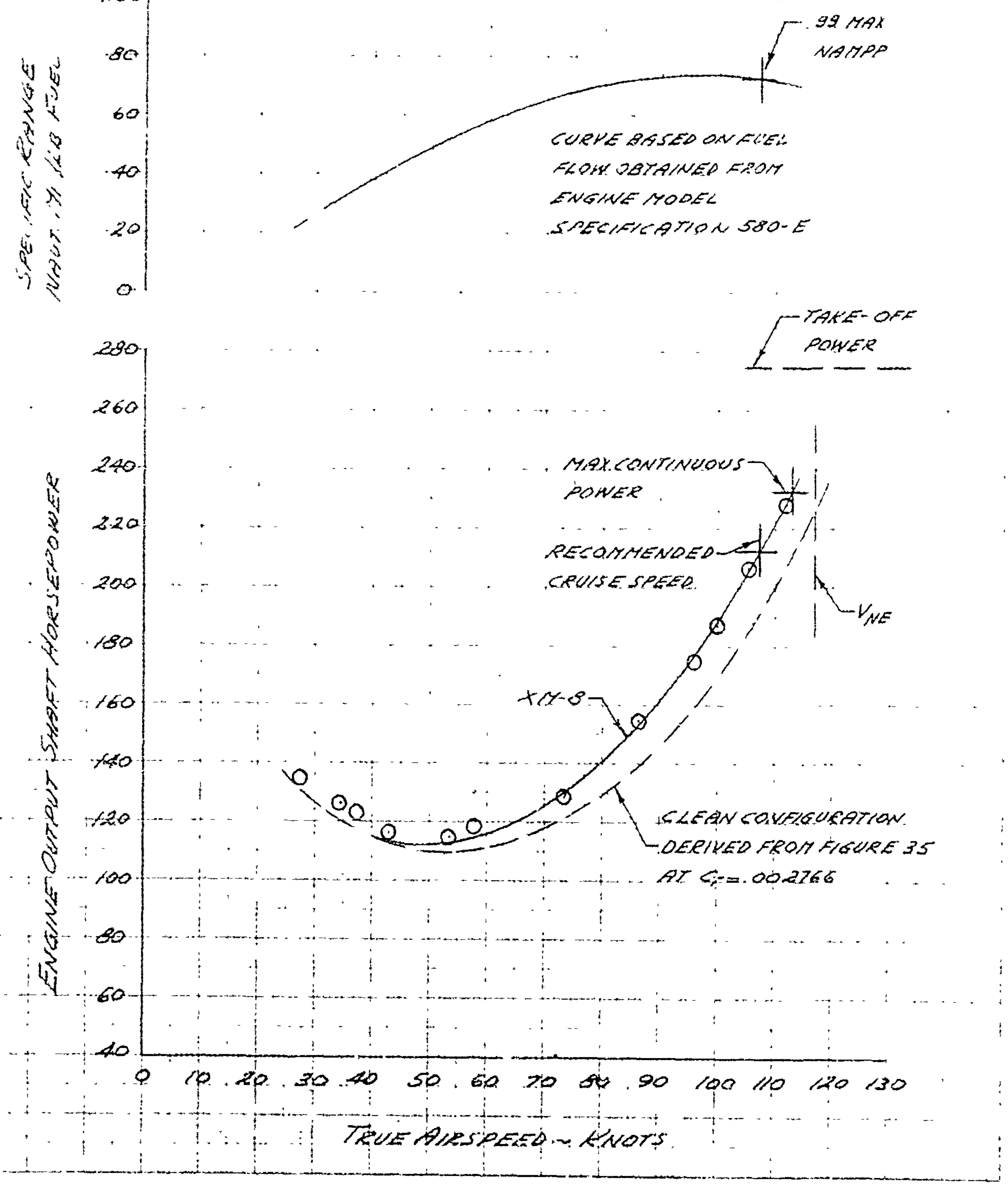


FIGURE NO. 53
AUTOROTATIONAL DESCENTS
OH-58A USA S/N 2-4207
ROTOR SPEED = 368 RPM
C.G. = MID (98.5)
CLEAN CONFIGURATION

SYM.	DENSITY ALT. ~ FEET ~	GROSS WEIGHT ~ LBS ~
○	5000	2460
⬠	10000	2150
△	10000	2915
▷	5000	2914
○	10000	2385
□	5000	2440
▽	10000	2470
◇	5000	2415
⬡	5000	2485
□	10000	2500

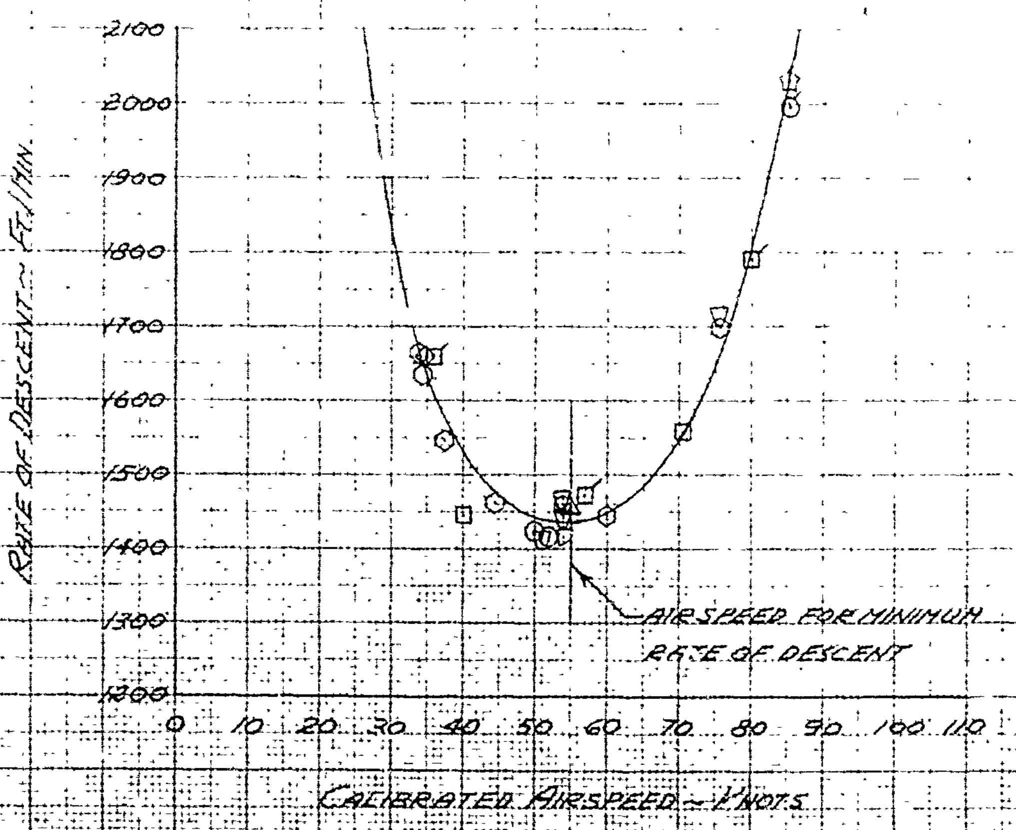


FIGURE No 54
AUTOROTATIONAL DESCENTS
OH-5A USA S/N 62-4207
AIRSPEED = 55 KNOTS (AS)
C.G. = MID (98.5)
CLEAN CONFIGURATION

SYM.	DENSITY ALT. ~ FEET ~	GROSS WEIGHT ~ LBS ~
○	10000	2435
□	10000	2530
△	10000	2450
⬡	10000	2440
◇	10000	2410
▽	10000	2545
○	10000	2615
□	10000	2525
○	5000	2450
◇	5000	2470
⬠	10000	2915
▽	5000	2475
D	10000	2275

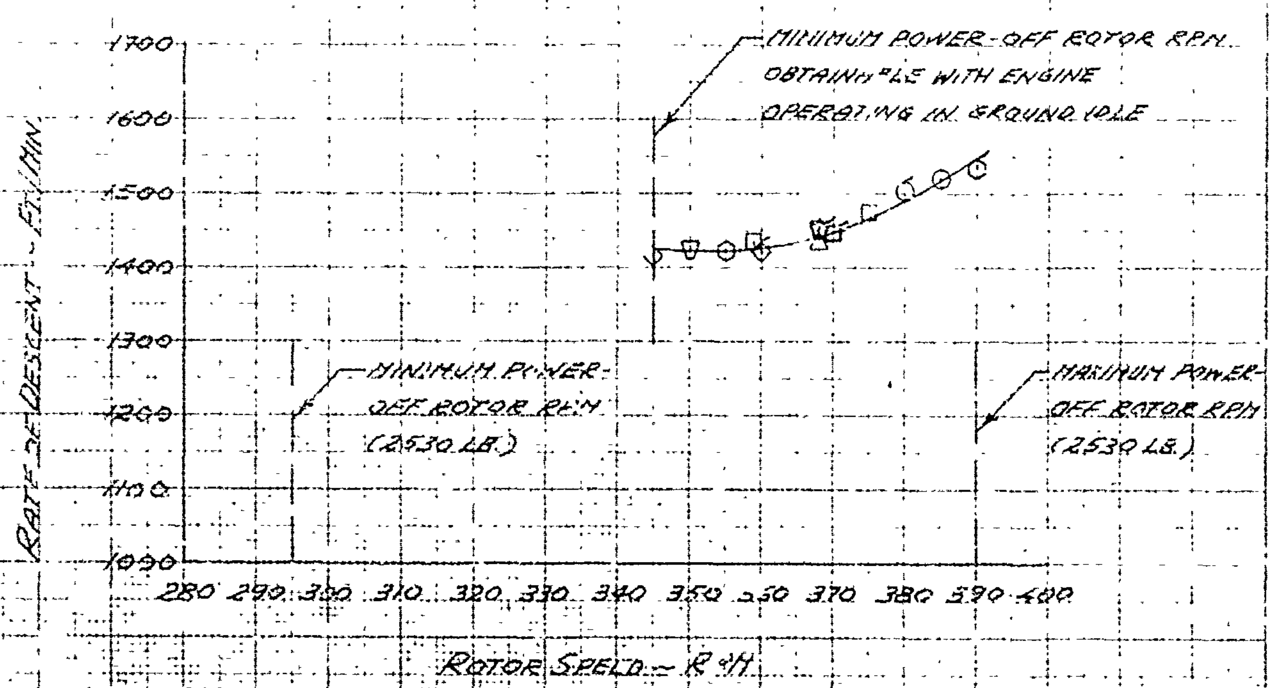


FIGURE No 55
AUTOROTATIONAL DESCENTS
OH-5A USA S/N 62-4207
ROTOR SPEED = 368 RPM
GROSS WEIGHT = 2530 LB.
ALTITUDE = 5000 FEET
C.G. = MID (98.5)
CLEAN CONFIGURATION

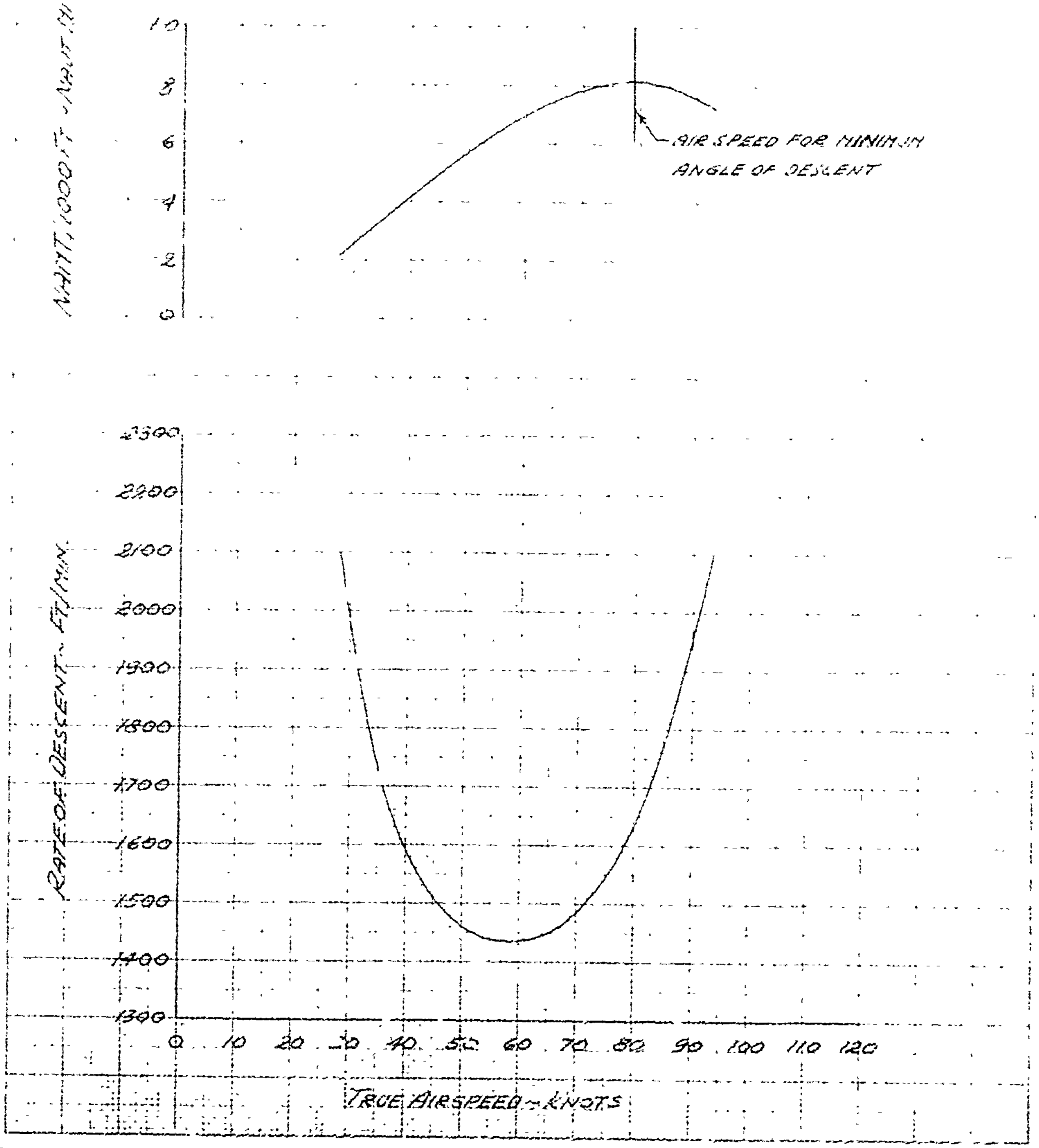


FIGURE NO. 56
SHAFT HORSEPOWER AVAILABLE
OH-5A USA S/N 62-4207
T-63-A-5
TAKE-OFF POWER

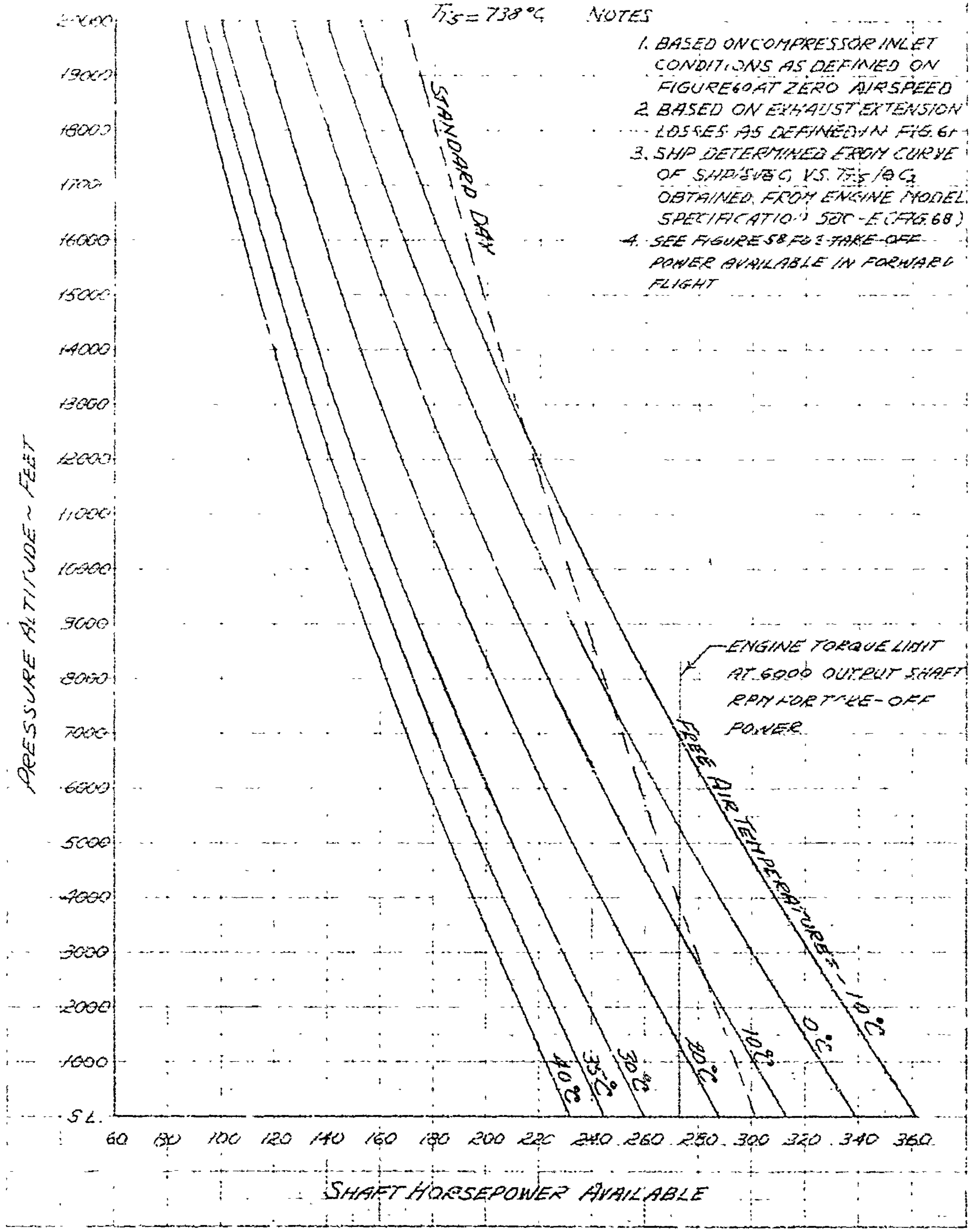


FIGURE No. 57

SHAFT HORSEPOWER AVAILABLE

OH-5A USA S/N 62-4207

T-63-A-5

MAXIMUM CONTINUOUS POWER

$T_{T5} = 693°C$

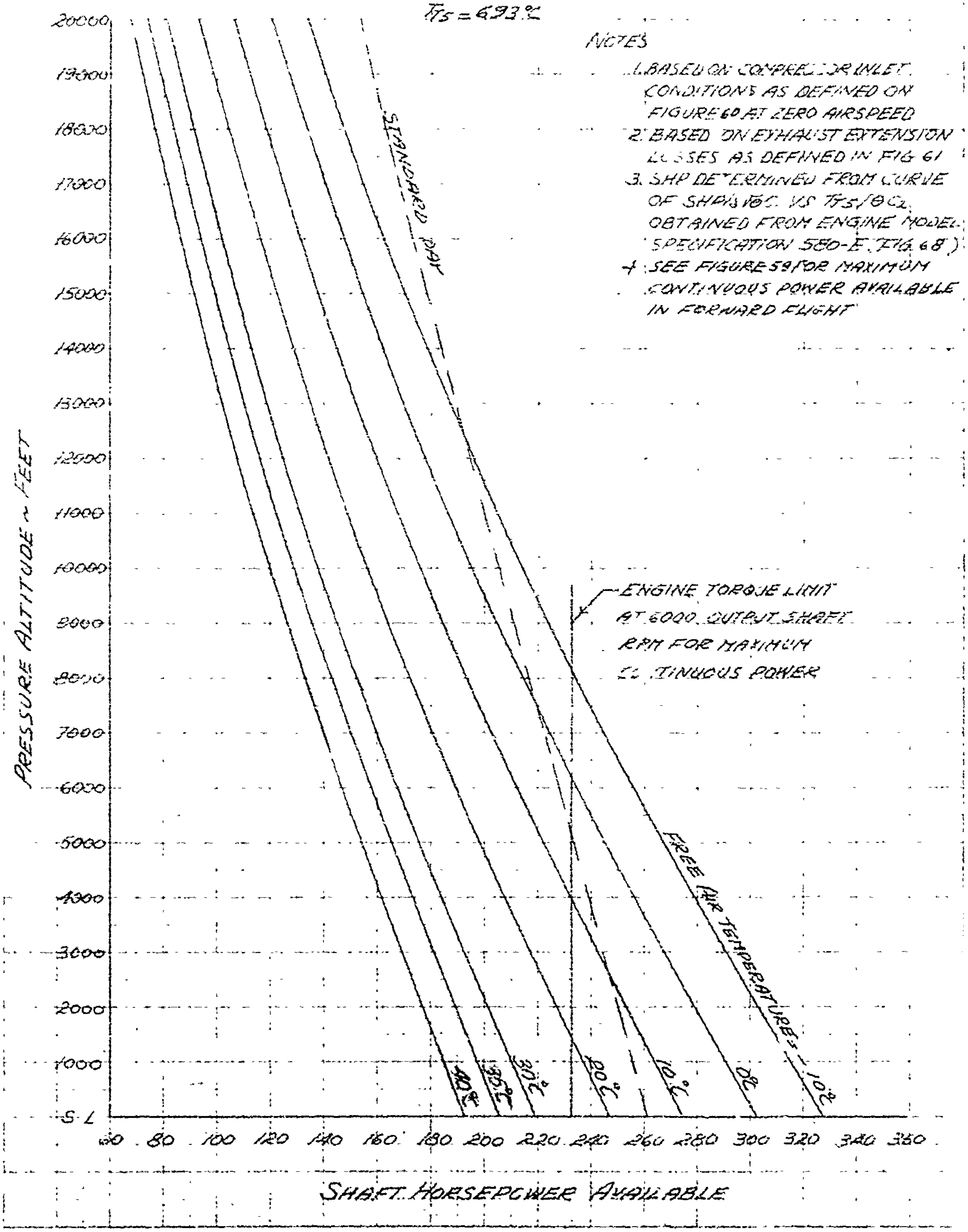


FIGURE No. 58
SHAFT HORSEPOWER AVAILABLE
OH-5A USA S/N 62-4207
T-63-A-5
TAKE-OFF POWER
$T_{T5} = 738°C$
US STANDARD DAY

NOTES

1. BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED ON FIGURE 60
2. BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 61
3. SHP DETERMINED FROM CURVE OF SHP/δ√θ VS T_{T5}/θ AS OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E (FIG 6.5)
4. SEE FIGURE 56 FOR SHP AVAILABLE DURING HOVER

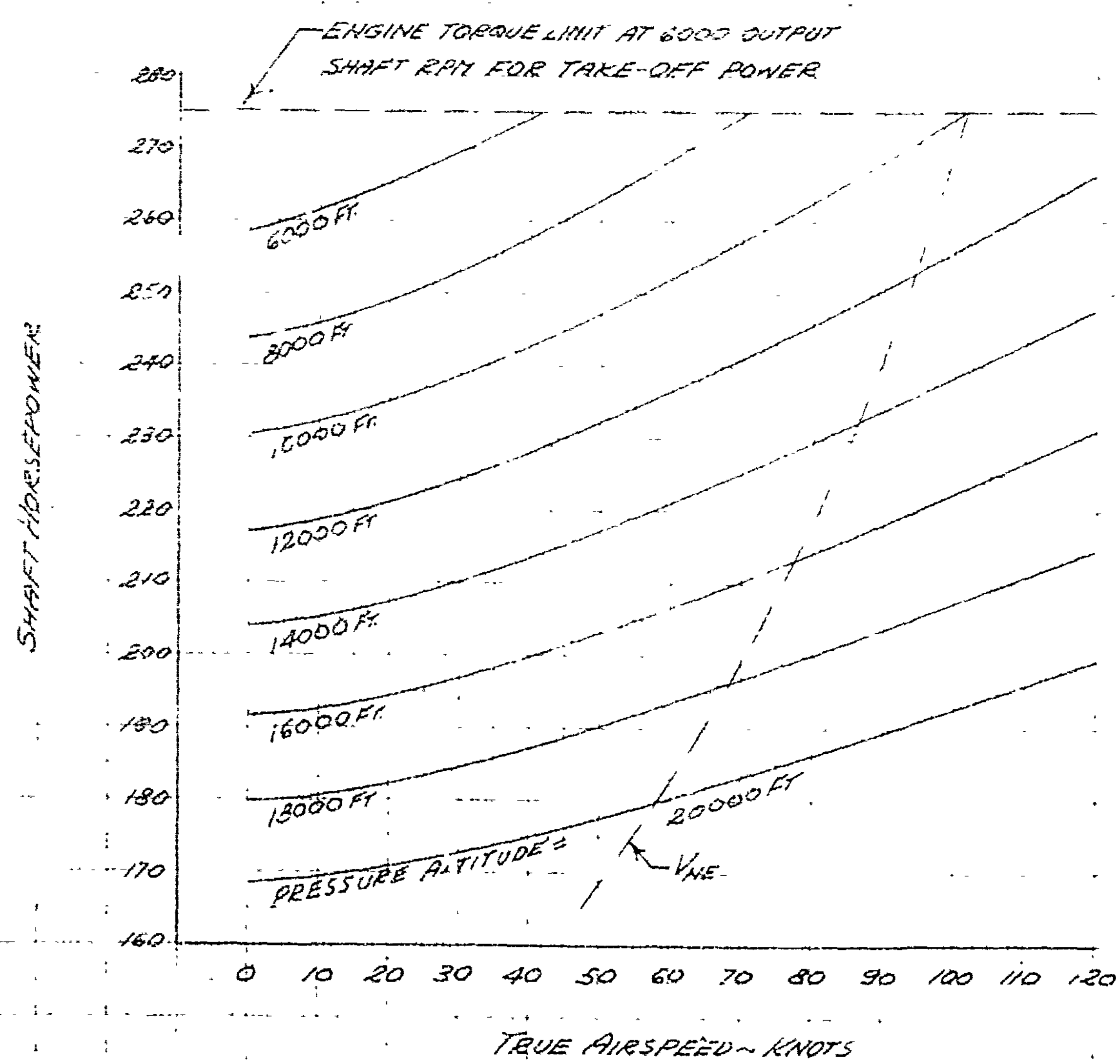


FIGURE No. 59
SHAFT HORSEPOWER AVAILABLE
OH-5A USA S/N 62-4201
T63-A-5
MAXIMUM CONTINUOUS POWER
T_{T5} = 693°C
US STANDARD DAY

NOTES
1 BASED ON COMPRESSOR INLET CONDITIONS AS DEFINED ON FIGURE 60
2 BASED ON EXHAUST EXTENSION LOSSES AS DEFINED IN FIGURE 61
3 SHP DETERMINED FROM CURVE OF SHP/δ√θC VS T_5/θC AS OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E (FIG 68)
4 SEE FIGURE 57 FOR SHP AVAILABLE DURING HOVER

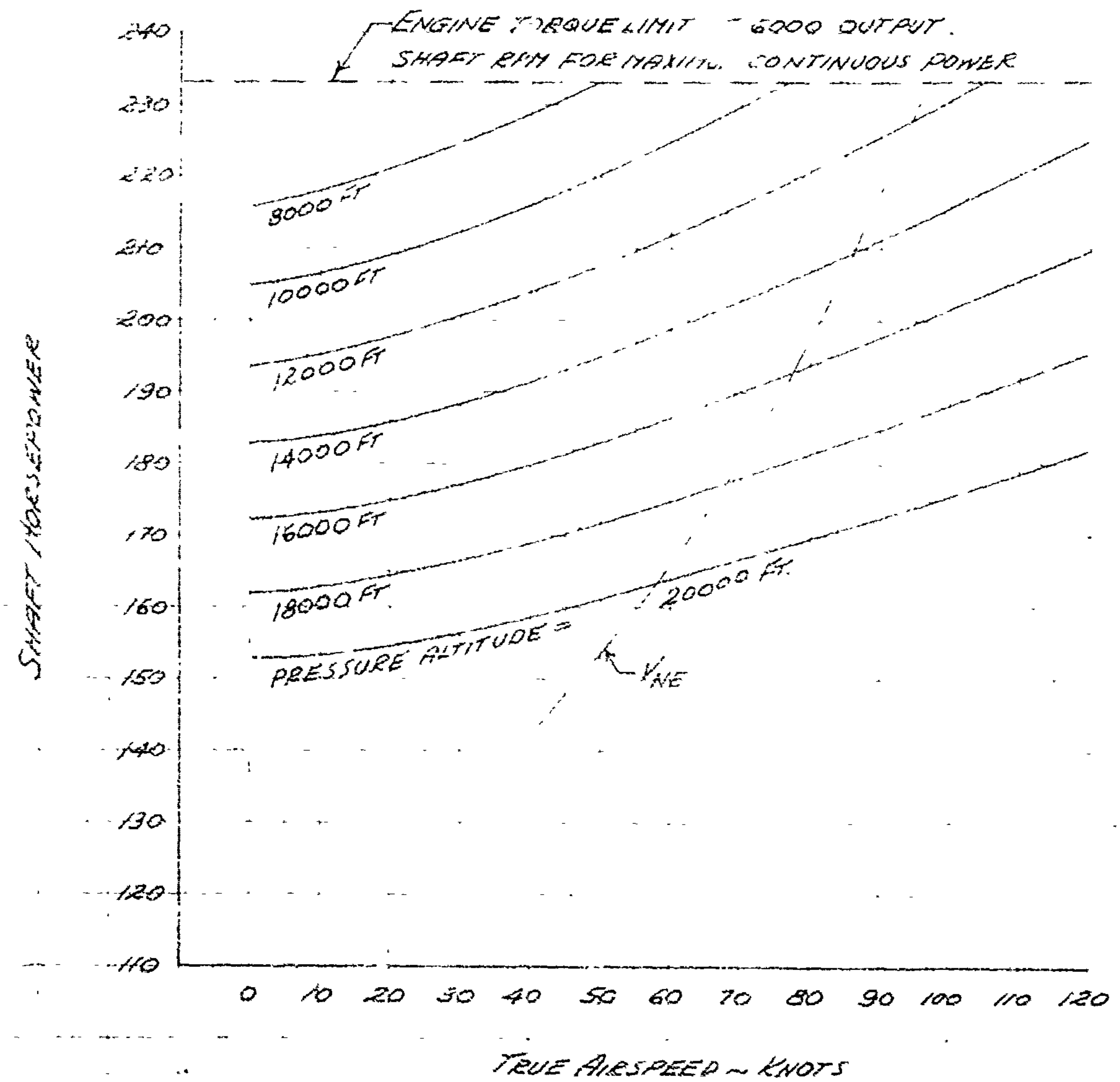


FIGURE No. 60
ENGINE CHARACTERISTICS
OH-5A USA S/N 62-4207
T63-A-5 S/N 400042

NOTE
POINTS OBTAINED FROM
READINGS TAKEN DURING
STABILIZED LEVEL FLIGHT

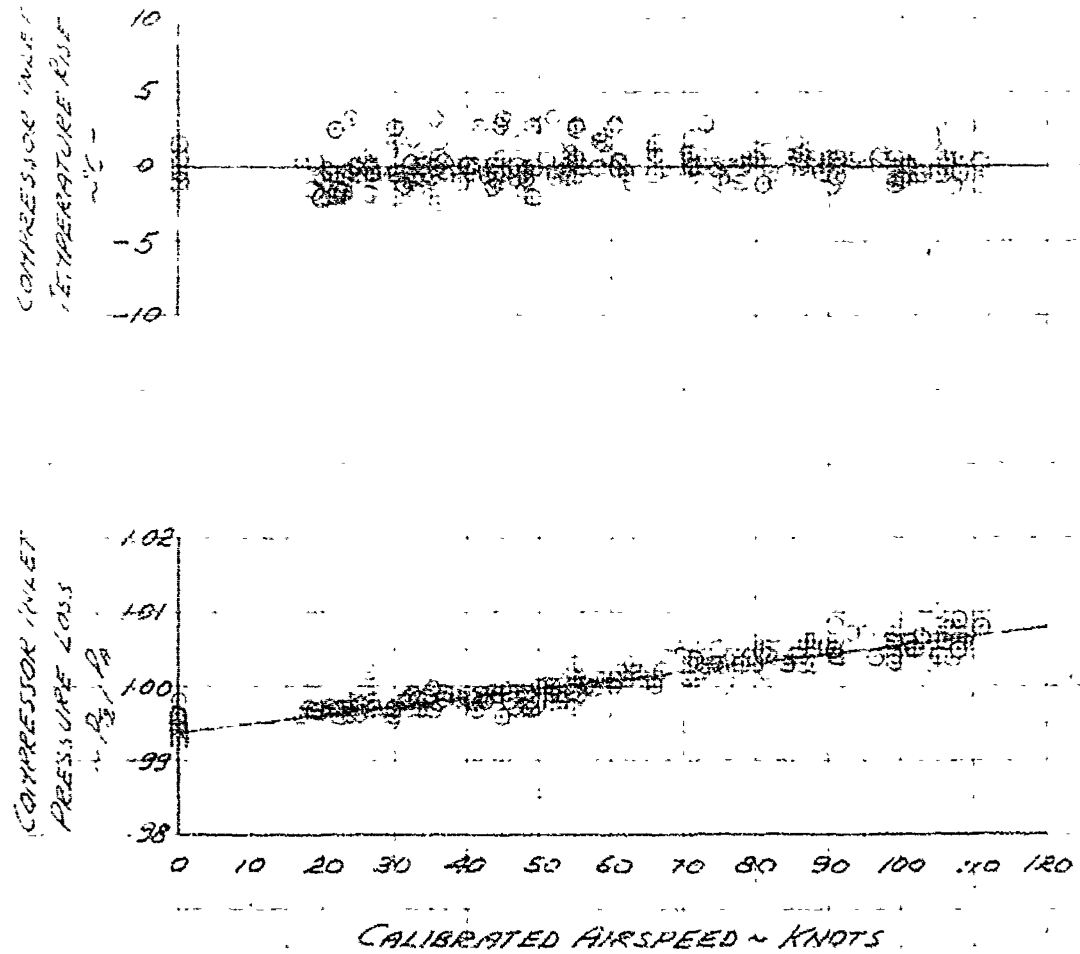


FIGURE NO 61

EFFECT OF HILLER EXHAUST EXTENSION ON T-63-A-5 EXHAUST STATIC PRESSURE

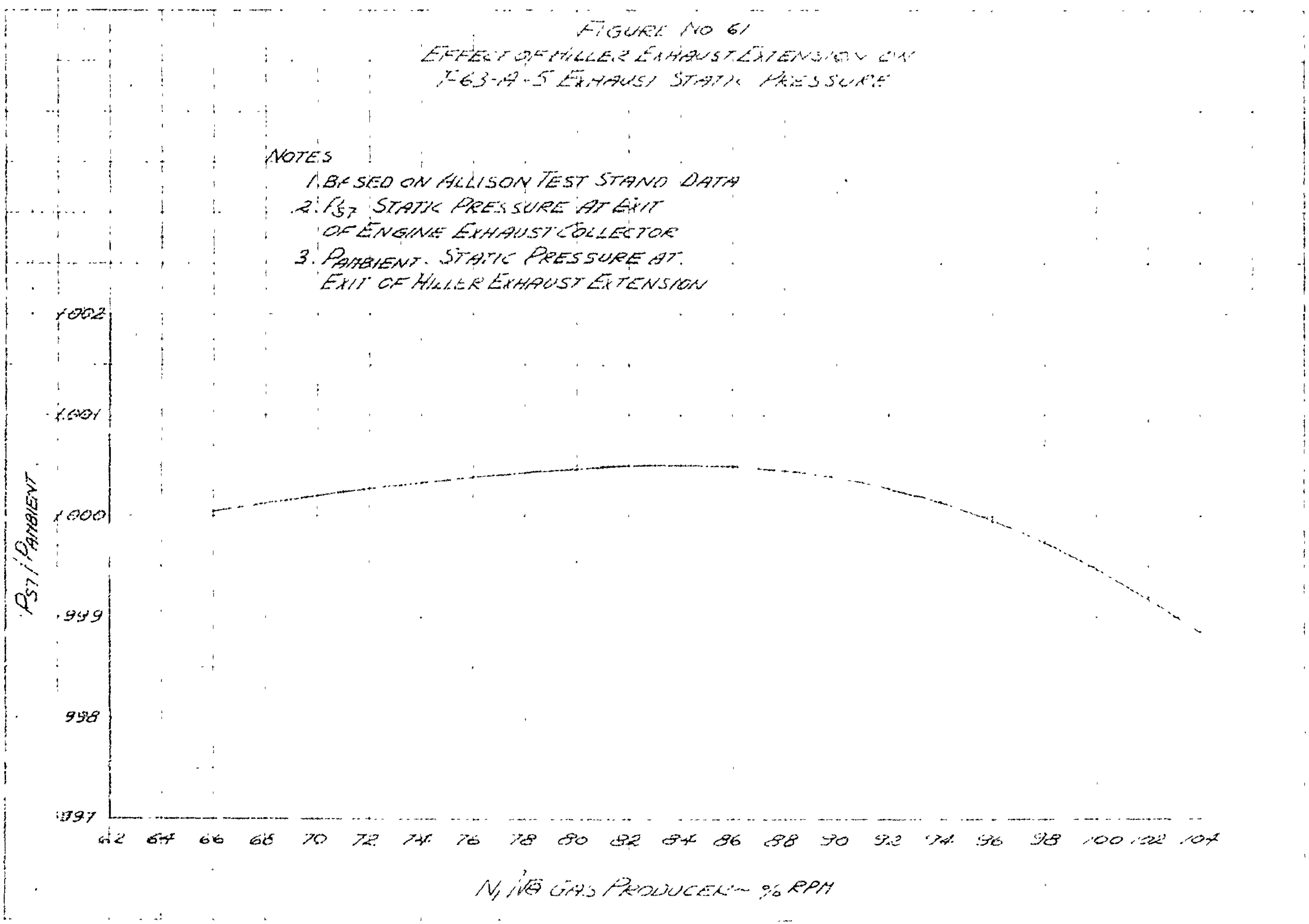


FIGURE NO. 62
ENGINE CHARACTERISTICS
T63-A-5 S/N 400042
OH-5A USA S/N 62-4207

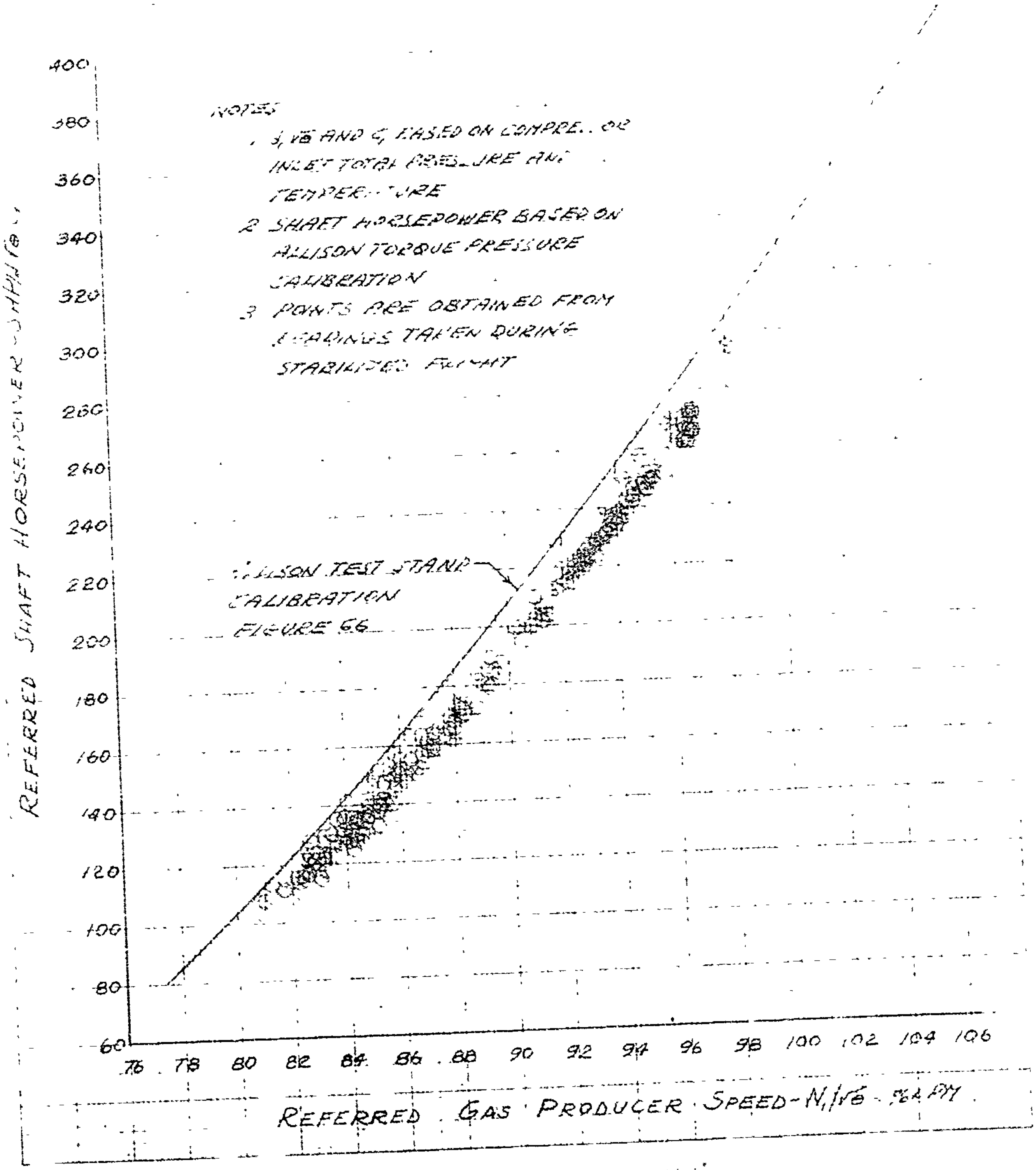


FIGURE No. 63
ENGINE CHARACTERISTICS
T63-A-5 S/N 400042
OH-5A USA S/N 62-4207

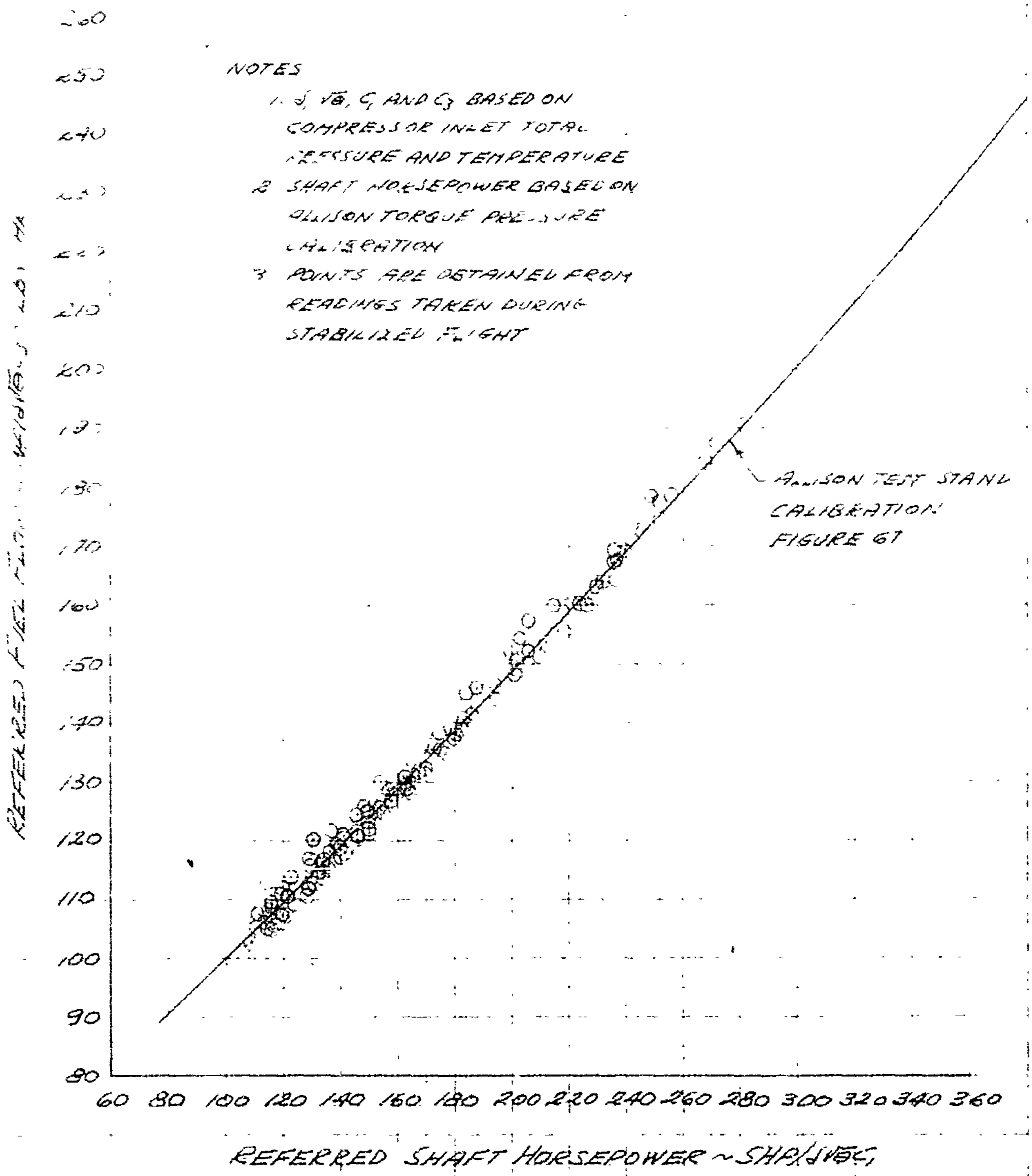


FIGURE NO. 64
ENGINE CHARACTERISTICS
T63-A-5 S/N 400042
OH-5A USA S/N 62-4201

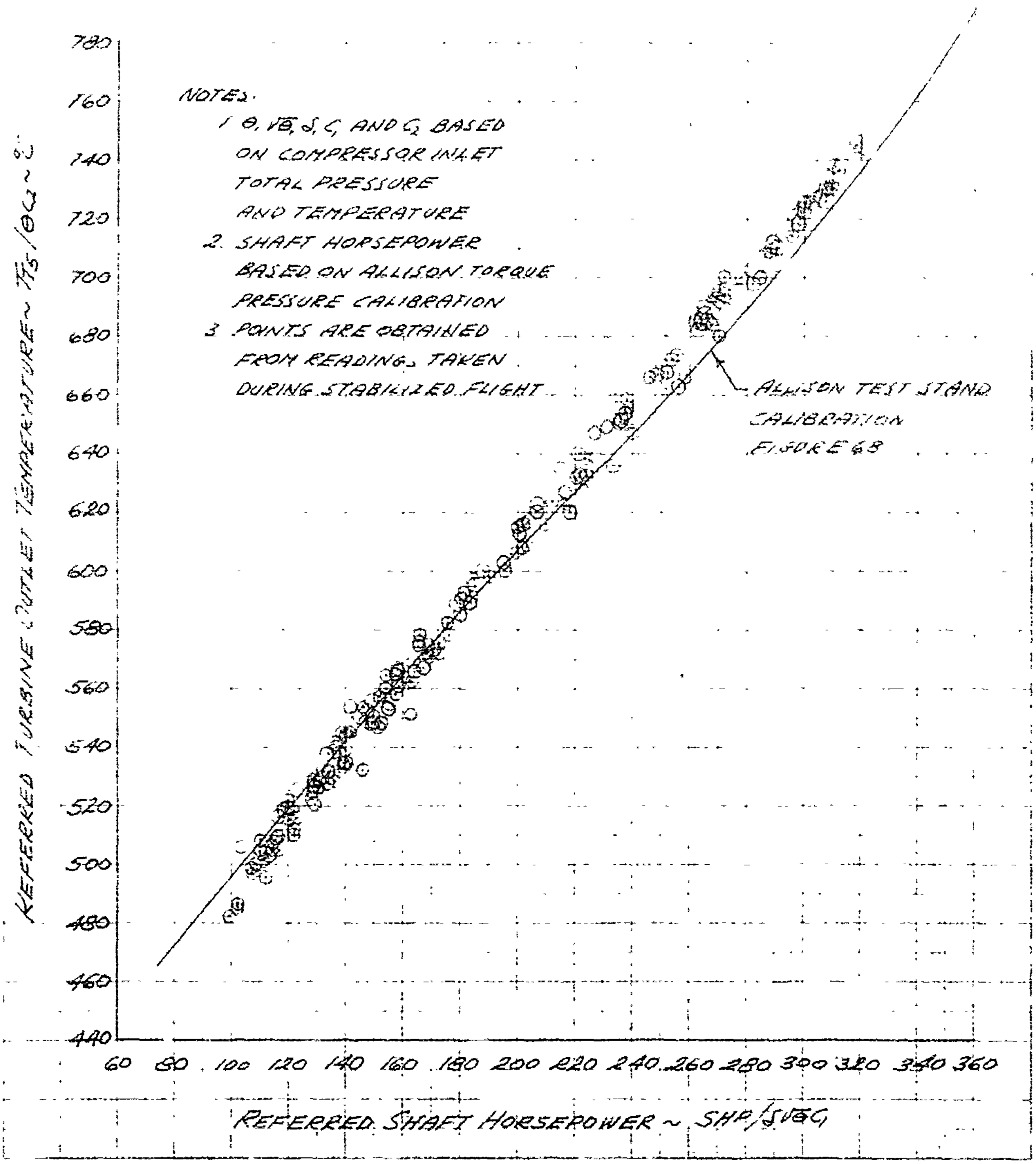
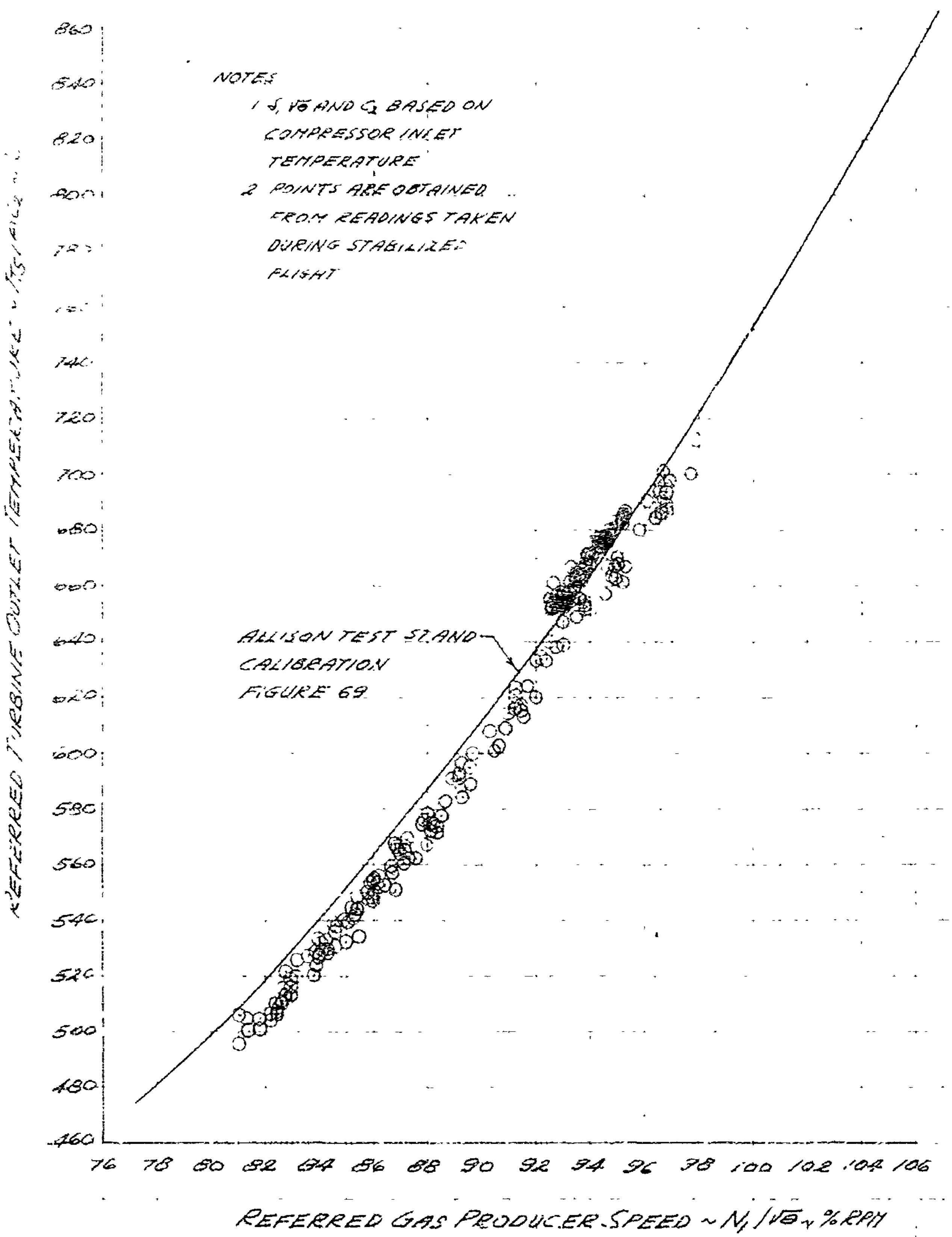


FIGURE NO. 65
ENGINE CHARACTERISTICS
T63-A-5 S/N 400042
OH-5A USA S/N 62-4207



FOR ... ONLY

FIGURE No. 66
ENGINE CHARACTERISTICS
T-63-A-5 S/N 400042
BASED ON ALLISON TEST
STAND CALIBRATION

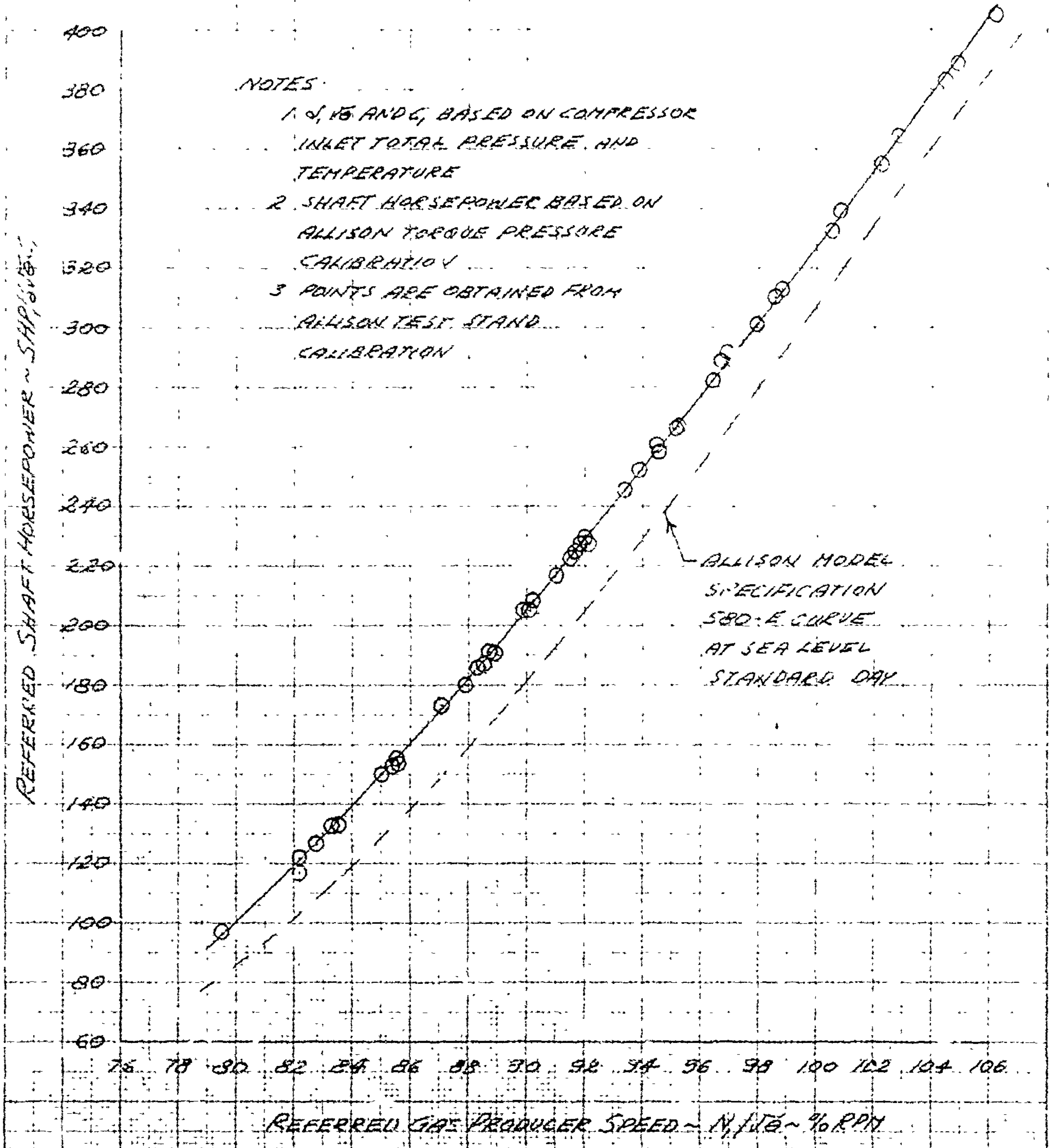
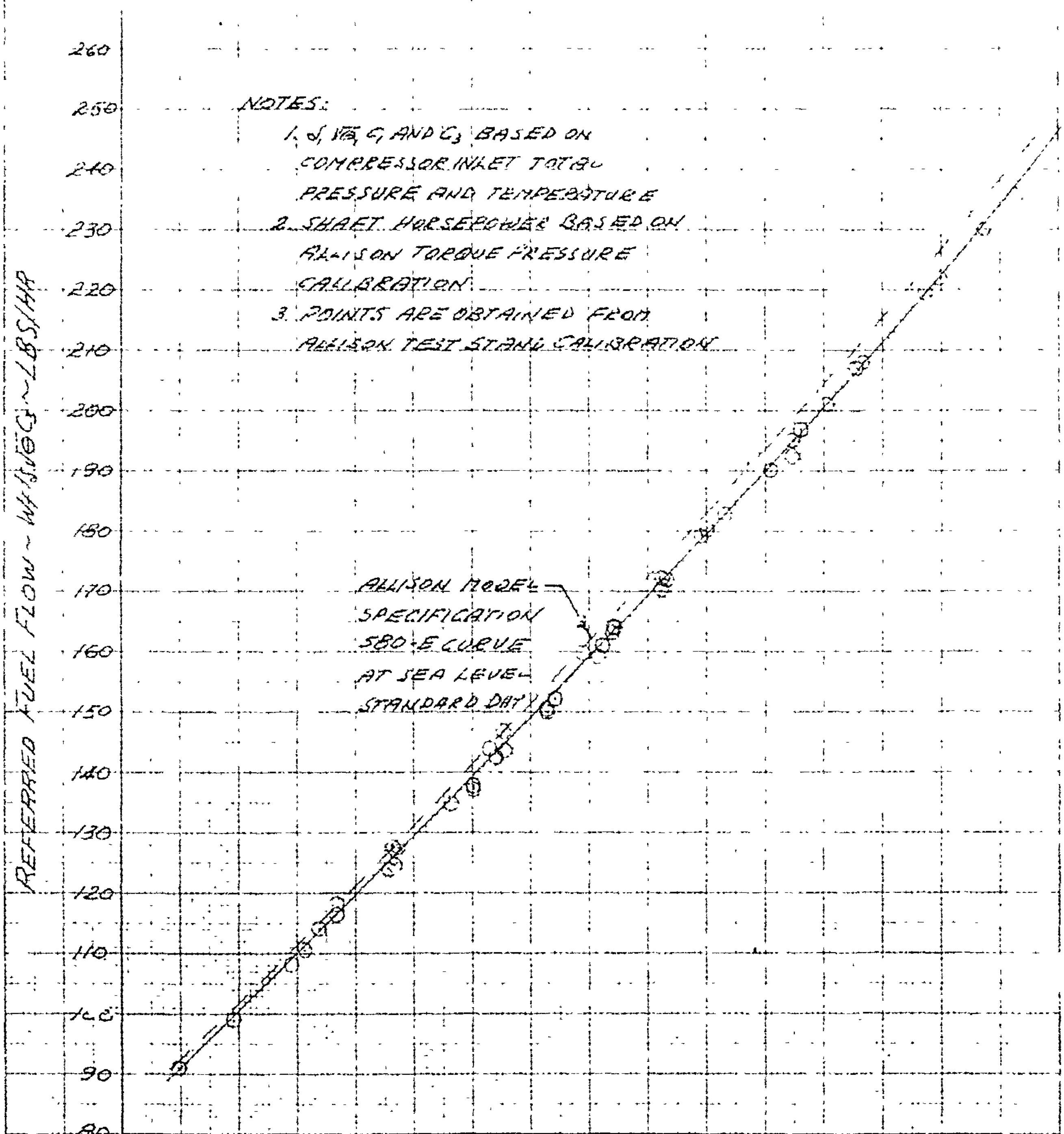


FIGURE NO. 67
ENGINE CHARACTERISTICS
T63-A-5 S/N 400042
BASED ON ALLISON TEST
STAND CALIBRATION



ONLY

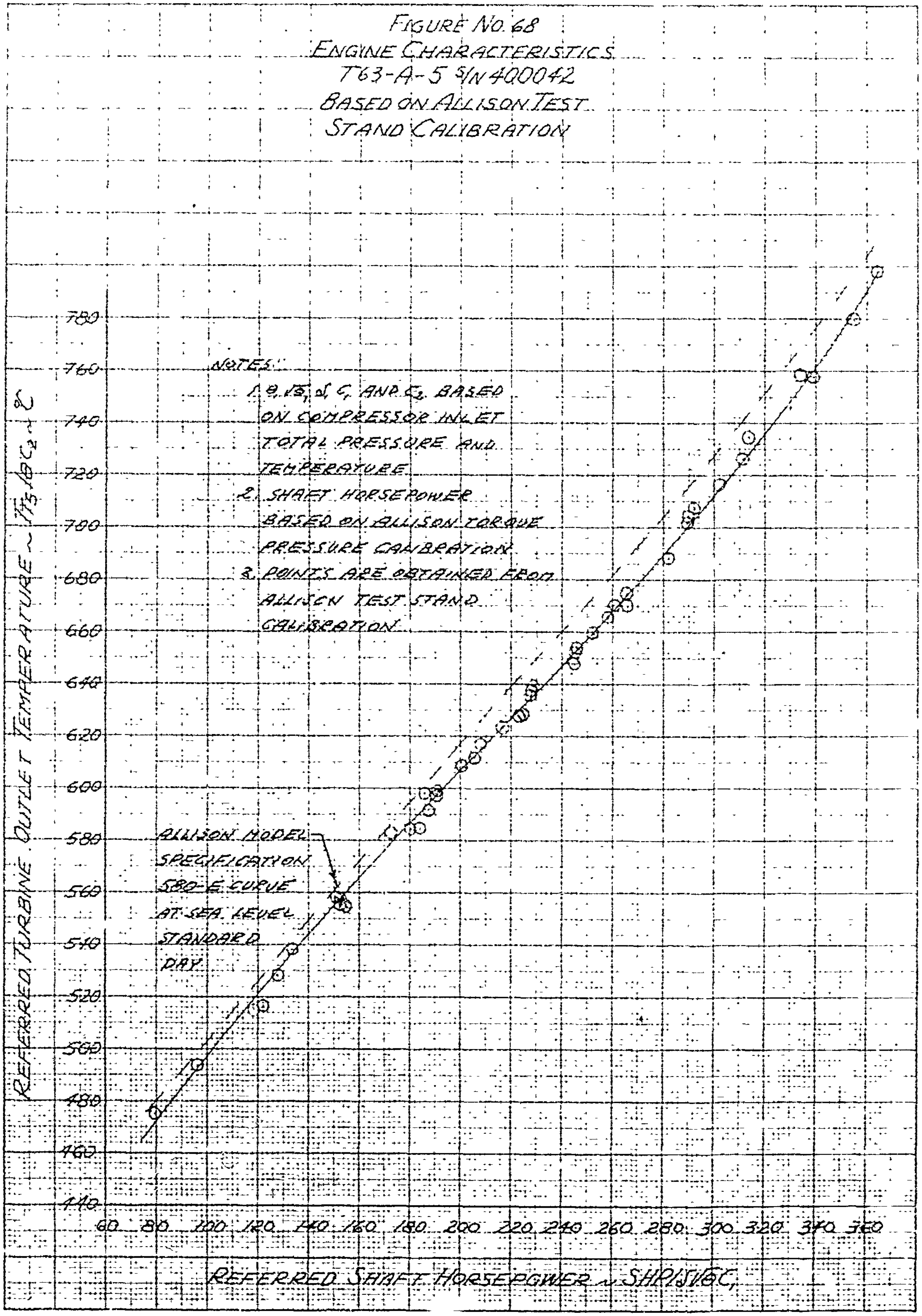
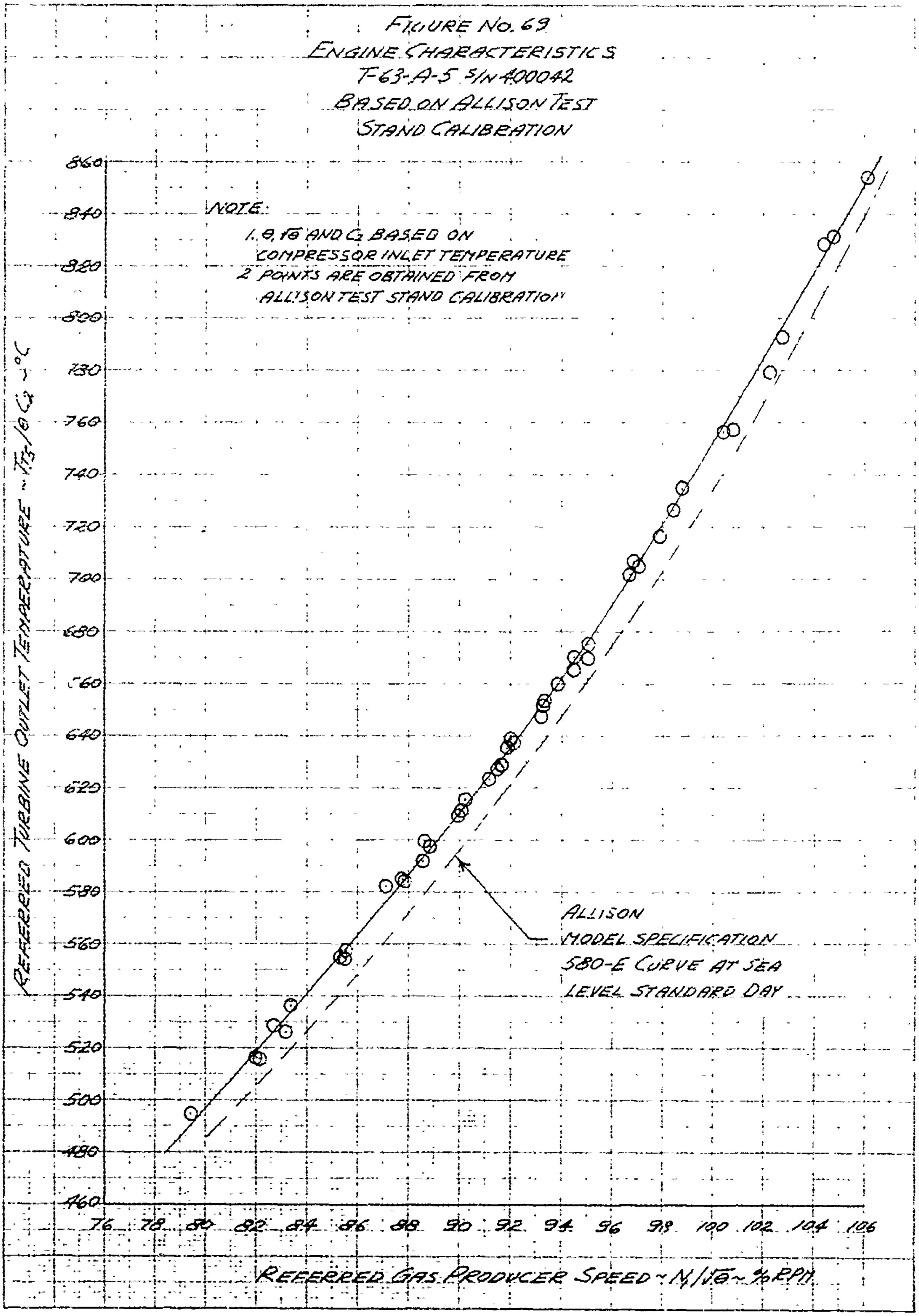
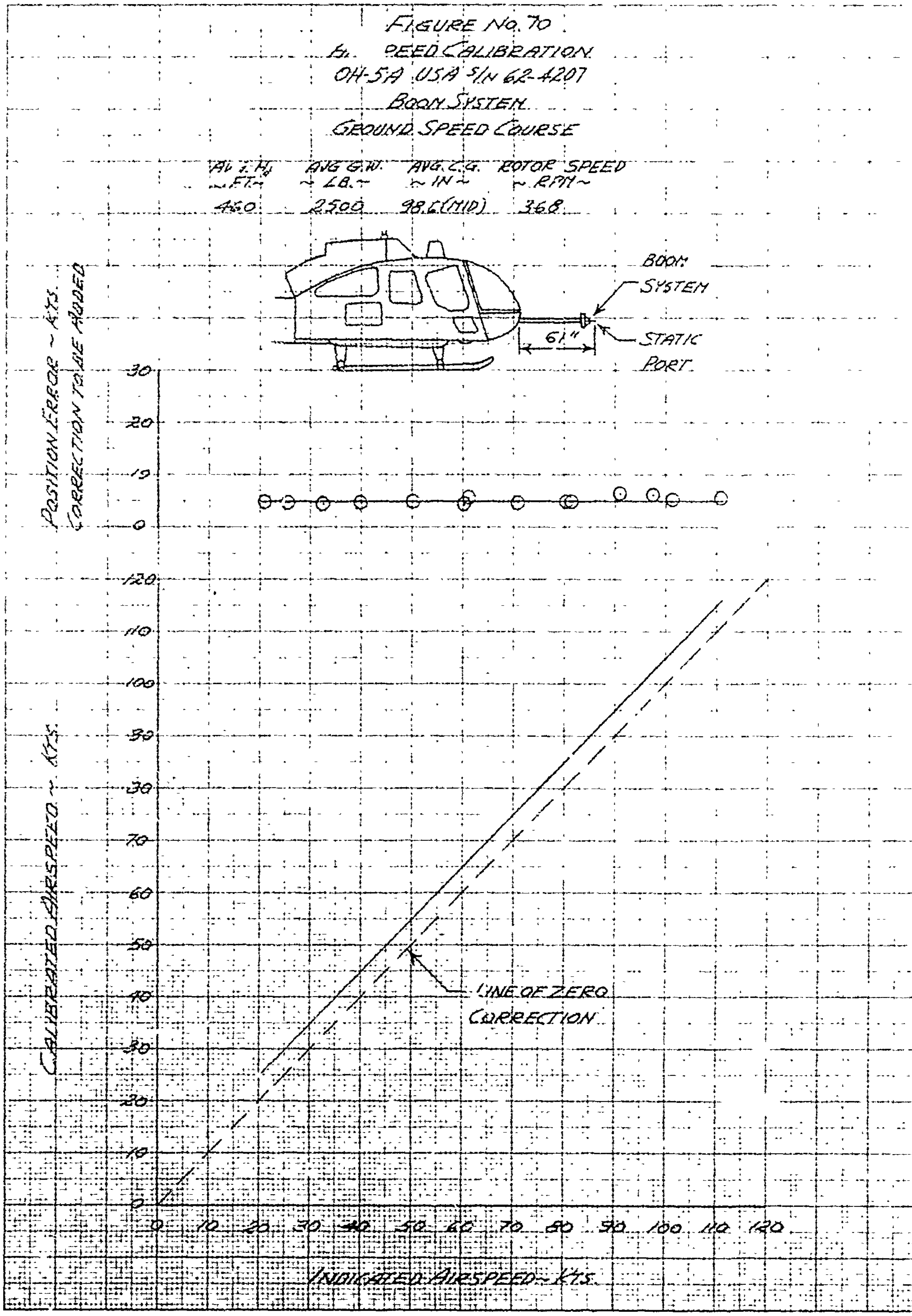


FIGURE NO. 68
ENGINE CHARACTERISTICS
T63-A-5 S/N 400042
BASED ON ALLISON TEST
STAND CALIBRATION



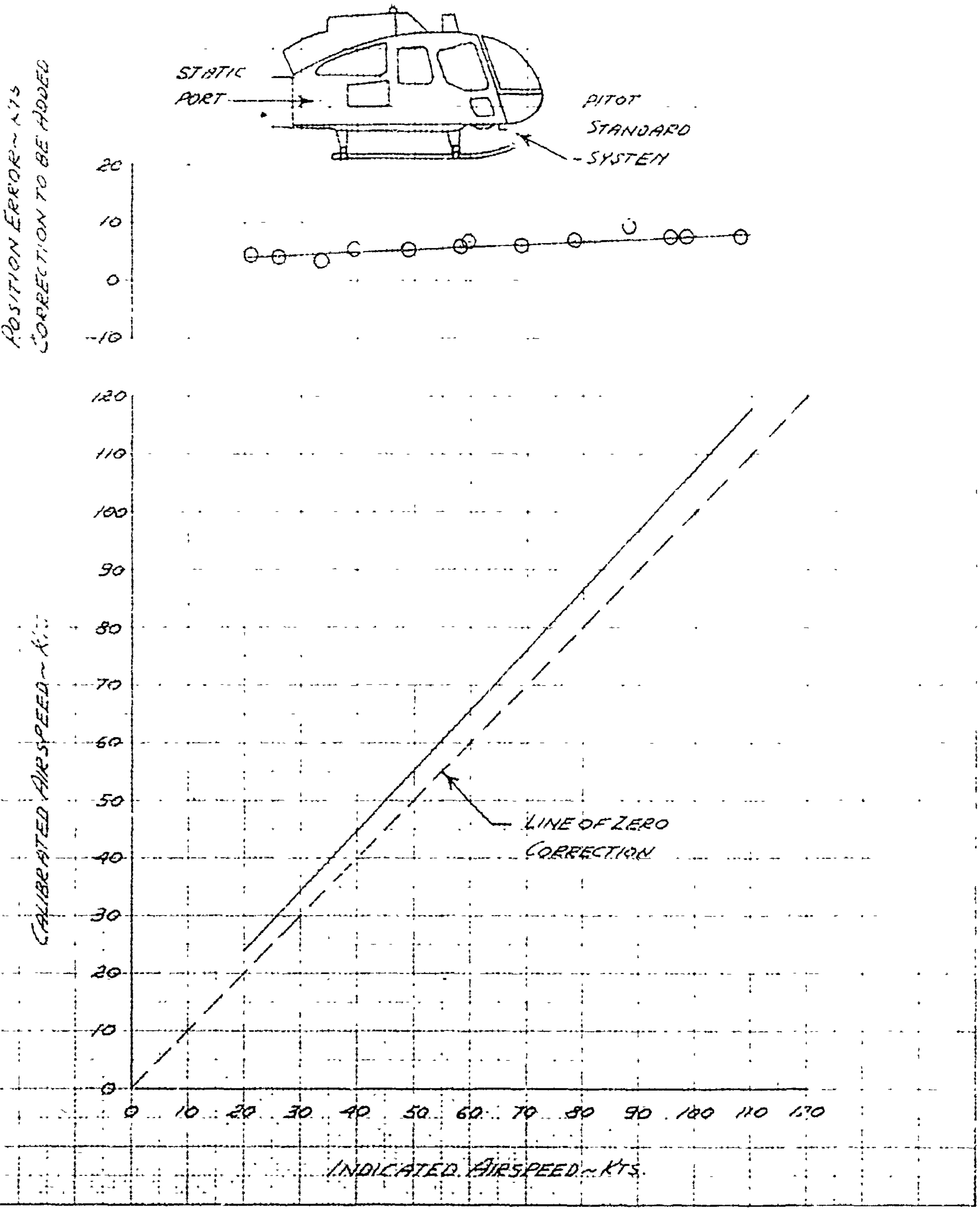
FOR OFFICIAL USE ONLY



FOR OFFICIAL USE ONLY

FIGURE NO. 71
AIRSPEED CALIBRATION
OH-5A USA S/N 62-4207
STANDARD SYSTEM
GROUND SPEED COURSE

AVG H_d ~ FT ~	AVG G.W. ~ LB. ~	AVG C.G. ~ IN. ~	ROTOR SPEED ~ RPM ~
460	2500	98.6 (MID)	368



APPENDIX II

GENERAL AIRCRAFT INFORMATION

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

Number of Main Rotor Blades	2
Main Rotor Diameter	35 ft (Tip Caps Add 5 in.)*
Main Rotor Blade Area	13.966 ft^2
Main Rotor Swept Area	983.73 ft^2
Main Rotor Disk Loading (Normal G.W.)	2.572 lb/ft^2
Main Rotor Solidity	.251
Number of Tail Rotor Blades	2
Tail Rotor Swept Area	28.26 ft^2
Tail Rotor Solidity	.108

* Rotor Diameter = 35 ft used for Performance Calculations

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-5A, 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 204 pound-feet torque at 6000 rpm (233 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 240 pound-feet torque (275 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 into which 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 compensation 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 marginally satisfactory throughout the test program.

There is no provision for emergency control in the 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

See Part I, Appendix II

2.2.4 Fuel System

See Part I, Appendix II

2.2.5 Control Systems

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 1476 pounds with the longitudinal C.G. located at Station 108.09.

In order to provide a basis for verifying whether or not the manufacturer's helicopter met the performance guarantees, the empty weight was defined as the weight of OH-5A, USA S/N 62-4207, as delivered to the U. S. Army Aviation Test Activity, with corrections for the increases in weight of Government furnished equipment, and corrections for items installed but not part of the contract empty weight (Reference Part I, Section 1, paragraph 1.1.1. With these corrections, the empty weight used was 1501 pounds (full oil and trapped fuel).

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

Normal Gross Weight = Empty weight + useful load

Empty Weight = 1501 pounds

Useful Load = Pilot - 200 pounds

Cargo - 400 pounds

Fuel as necessary to meet normal gross weight of Model Specification, paragraph 3.2.1 - 429 pounds

OH-5A Model Specification, paragraph 3.2.1 states that the normal gross weight performance guarantee weight shall not exceed 2410 pounds ± 5 percent; therefore, helicopter performance was calculated at 2530 pounds (2410 + 5 percent).

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

Copilot flight controls	-	10.0 pounds
Anti-collision light	-	2.66 pounds
Ground handling wheels		27.5 pounds
XM-7 armament system (less ammunition)		140.0 pounds
XM-8 armament system (less ammunition)		142.0 pounds

After installation of test instrumentation, the helicopter was again weighed in the same configuration as before with the exception of installed instrumentation. The basic weight (full oil and trapped fuel) was 1792 pounds with the longitudinal C.G. located at Station 103.14.

5. Test instrumentation

The test instrumentation used during this evaluation was supplied, installed and maintained by the Logistics Division of the U. S. Army Aviation Test Activity. Calibration of the instrumentation was accomplished by the Instrumentation Branch of the Air Force Flight Test Center; Lockheed Aircraft Corporation; Douglas Aircraft Missle and Space Division; and the Logistics 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. 70, Section 3, Appendix I. Sensitive instrumentation was installed prior to initiation of the test flights to measure the following parameters.

Pilot-Engineer Panel

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 (Tt_5)

Compressor Inlet Total Temperature

Compressor Inlet Total Pressure

Exhaust Gas Static Pressure

Cockpit Absolute Pressure

Fuel Flow (Stepper Motor)

Total Fuel Used

Photo Panel Frame Counter

Oscillograph Record Counter

Photo Panel

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 Event

Engineer 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 existance 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. Engine Operation

During the performance evaluation of the OH-5A helicopter the T63-A-5 engine operating characteristics were found to be essentially the same as reported in Section 1, paragraph 1.1m and Part I, Section 2, paragraph 2.8.5.

The T63-A-5 engines used during this evaluation revealed a new problem in the form of excessive dirt accumulation in the compressor section of the engine. This accumulation of dirt was under the centrifugal compressor shroud. Power deterioration was evidenced by an increase in acceleration time or by the inability to accelerate to ground idle. To clean the compressor it was necessary to remove the compressor section from the engine and disassemble the axial compressor case. The dirt buildup was then removed with a wire brush. One compressor was modified by the engine manufacturer to relieve this problem. Ths modified compressor accumulated dirt at an accelerated rate. The unmodified compressor required cleaning after approximately 50 hours of engine operating time whereas the modified compressor required cleaning after only 25 hours of engine operating time.

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	$\frac{lb\text{-}sec^2}{ft^4}$