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U.S. ARMY

TEST & EVALUATION COMMAND

USATECOM PROJECT NO. 4-3-0250-11/12/13

PART TWO OF TWO PARTS

REPORT OF THE ENGINEERING FLIGHT TEST PERFORMANCE PHASE OF THE OH-4A HELICOPTER, UNARMED (CLEAN) and ARMED WITH THE XM-7 or XM-8 WEAPON SUBSYSTEM

AUGUST 1964

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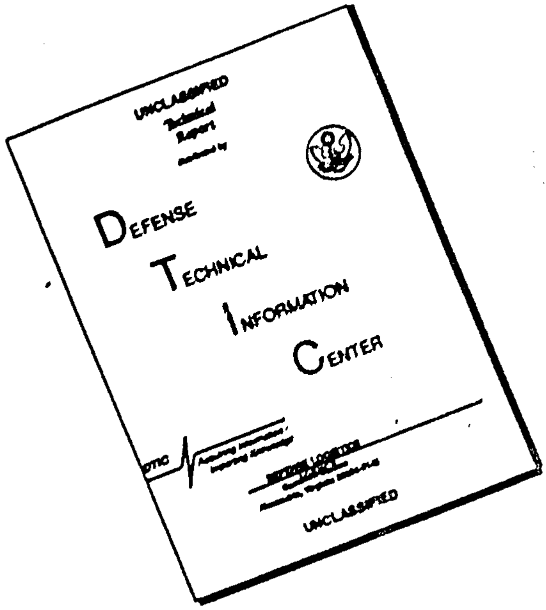

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U. S. ARMY AVIATION TEST ACTIVITY

(6) PART TWO OF TWO PARTS

REPORT
OF THE ENGINEERING FLIGHT TEST
--PERFORMANCE PHASE--
OF THE
OH-4A HELICOPTER, UNARMED (CLEAN) AND ARMED
WITH THE XM-7 OR XM-8 WEAPON SUBSYSTEM. Part II.

(16) USATECOM [redacted]-4-3-0250-11/[redacted]

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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.

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ABSTRACT

Performance flight tests were conducted on the OH-4A to determine the compliance with the Light Observation Helicopter (LOH) Military Characteristics and to check the contractual guarantees outlined in the OH-4A 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 of 121 flights were conducted for 73:45 productive flight hours. These tests were accomplished during the period of 13 March to 30 June 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-4A met the contractual performance guarantee for maximum airspeed.

The OH-4A did not meet the contractual performance guarantees for endurance or for 35 degrees centigrade (C) out-of-ground effect (OGE) hovering ceiling.

The OH-4A met the Military Characteristic requirement for 3 hours endurance with a 400 pound payload.

The OH-4A 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-4A performance was summarized using power available as defined by Engine Model Specification No. 580-E.

Takeoff performance was satisfactory for all conditions that allowed a hovering skid height of 2 feet or more. Hovering performance was satisfactory at low altitude standard day conditions but deteriorated rapidly with increasing ambient temperature and increasing altitude. Climb and autorotation performance was satisfactory for a helicopter of this

power and weight class. Blade stall was not encountered. Vibration levels were exceptionally low.

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-4A

PHOTO 2 - OH-4A

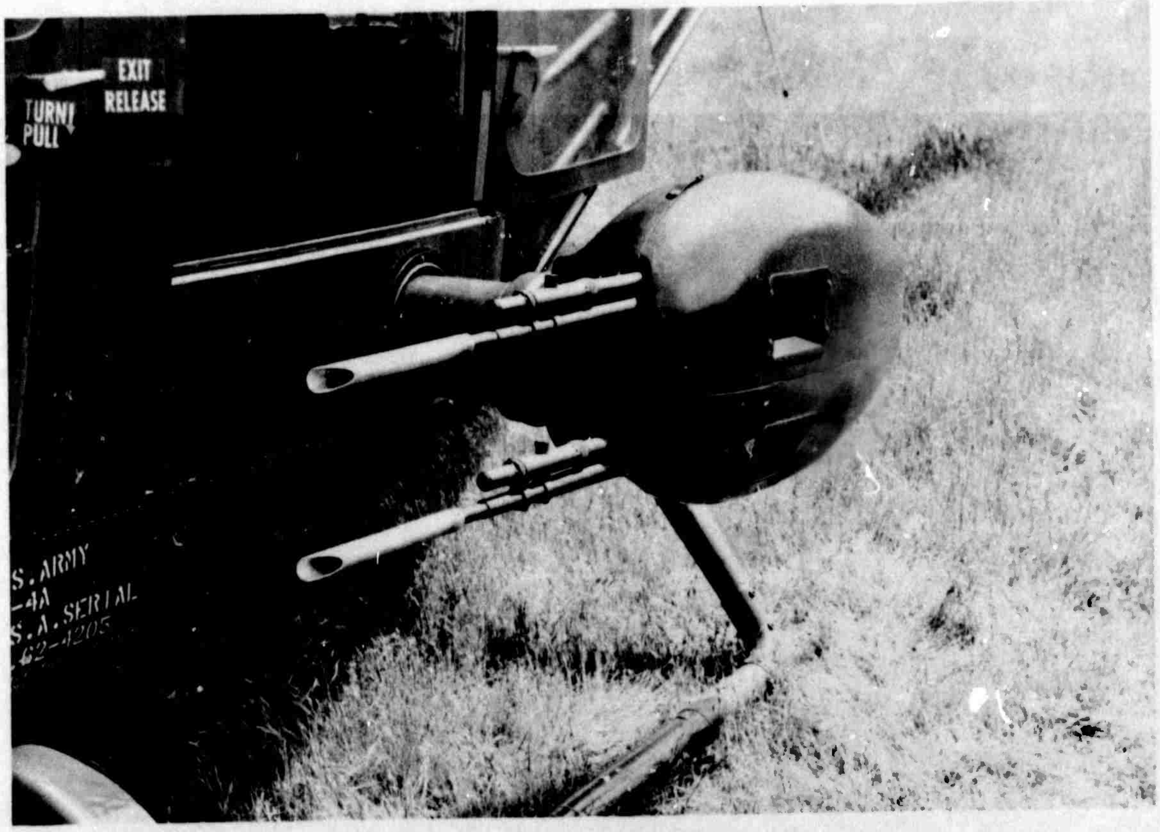


PHOTO 3 – XM-7 ARMAMENT KIT

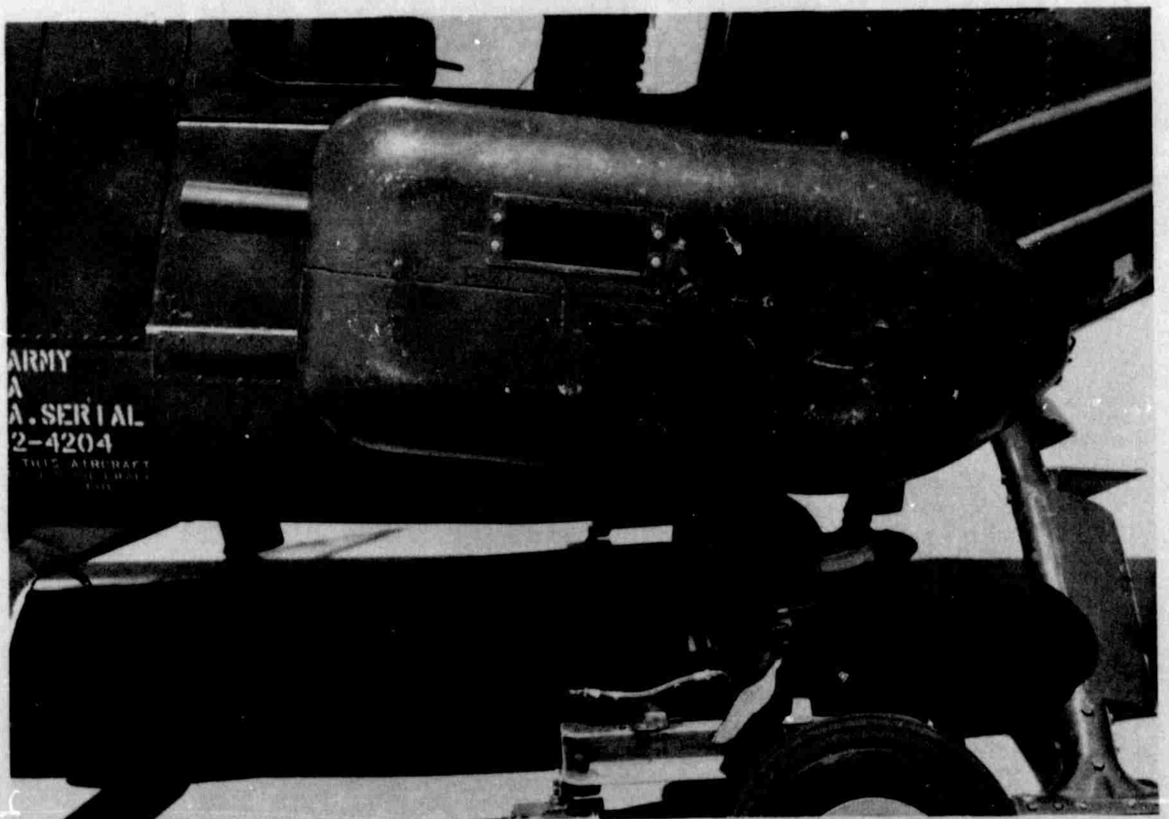


PHOTO 4 – XM-8 ARMAMENT KIT

SECTION 1 - GENERAL

1.1 REFERENCES

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

k. Preliminary Report, "Flight Evaluation of the T-63-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.

l. Model Specification, Light Observation Helicopter, OH-4, 15 September 1961; Revision No. 1, 8 December 1961.

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

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

o. 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."

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-4A to (a) confirm contractor compliance with the approved Army Military Characteristics for an unarmed (clean) and armed OH-4A 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 1, Section 1; Part I, Section 3, Appendix 11; and Part II, Section 3, Appendix 11.

1.6 BACKGROUND

See Part 1, Section 1.

1.7 FINDINGS

The takeoff performance of the OH-4A was satisfactory for all gross weights and ambient atmospheric conditions that allowed a hovering skid height of 2 feet. Under conditions that allowed a hovering skid height of approximately 2 to 2-1/2 feet, the minimum distance to clear a 50 foot obstacle was produced by a "level acceleration from a 2 foot hover" takeoff technique. Under conditions that allowed a hovering skid height greater than 2-1/2 feet, shorter takeoff distances were obtained using a simultaneous "climb and accelerate from light on skids" technique. Greater pilot proficiency was required to execute the "climb and accelerate" technique with consistent results. Engine rpm governing deficiencies required that the pilot monitor engine operation closely during takeoffs while using either technique.

Hovering performance was adequate based on power data obtained from the T63-A-5 Model Specification 580-E at lower altitudes on a standard day. Performance deteriorated rapidly with increasing ambient temperature and altitude. Based on power data from the T63-A Model Specification 580-A, the hot day (+35 degrees centigrade (C)), normal gross weight hovering performance guarantee was not met. The out-of-ground effect (OGE) hovering ceiling guaranteed was 4300 feet ± 10 percent. The actual OGE hovering ceiling at the gross weight and ambient conditions specified was 825 feet pressure altitude. The Military Characteristics, Light Observation Aircraft (See Part I, Section 1, paragraph 1.1a) required an OGE hovering ceiling of 6000 feet pressure altitude on a 35 degree C day at normal gross weight (2573 pounds). The maximum gross weight at which the OH-4A could hover OGE at these ambient conditions was 2085 pounds. The Military Characteristics also required OGE hover at sea level on a standard day at overload gross weight (2900 pounds). The maximum gross weight at which the OH-4A would hover OGE at sea level on a standard day was 2730 pounds.

The performance guarantees and Military Characteristics requirements were based on power available as defined in Engine Model Specification 580-A. The hovering performance summaries were based on Engine Model Specification 580-E. This revised model specification results in better standard day performance below critical altitude due to a higher torque limit but more rapid deterioration of power available with increasing ambient temperature. The overload gross weight (2900 pounds) OGE hovering ceiling was 2200 feet on a standard day. The maximum gross weight for hovering OGE on a 35 degree C day at 6000 feet was 1986 pounds. The OH-4A would not hover OGE at sea level on a 35 degree C day at normal gross weight (2573 pounds).

The climb performance of the OH-4A was satisfactory. With a climb start gross weight of 2573 pounds at sea level, the service ceiling was 20,150 feet on a standard day. Sea level rate of climb was 1100 feet per minute at maximum continuous power and 1550 feet per minute at takeoff power. With a climb start gross weight of 2900 pounds at sea level, the service ceiling was 15,600 feet on a standard day. Sea level rate of climb was 810 feet per minute at maximum continuous power and 1250 feet per minute at takeoff power.

During level flight, the maximum airspeed performance guarantee was met. The maximum airspeed guaranteed was 100 knots ± 10 percent at normal gross weight (2573 pounds) on a sea level standard day at maximum continuous power. Based on Engine Model Specification 580-A, the maximum true airspeed (TAS) was 96.5 knots at these conditions. The Military Characteristics, Light Observation Aircraft, required 110 knots at the same conditions. This requirement was not met by a margin of approximately 12.3 percent. The guaranteed endurance mission of 3.0 hours ± 10 percent was not met by a margin of approximately 0.37 hours (13.6 percent). The Military Characteristics required a 400 pound payload to be carried for 3 hours. At the airspeed for minimum power required in level flight, this requirement was met by a margin of approximately 0.18 hours (6 percent). The Military Characteristics also required 3 hours endurance at 85 percent cruise power. This requirement was not met by a margin of approximately 0.45 hours (15 percent).

The equivalent flat plate area of the XM-7 armament system was found to be 2.46 square feet and that of the XM-8 was 1.57 square feet. This reduced the sea level standard day at normal gross weight maximum speed from 111.5 KTAS to 105 KTAS with the XM-7 and to 107.5 KTAS with the XM-8 at takeoff power available as defined by Model Specification No. 580-E. Center of gravity (C.G.) did not significantly affect level flight performance. Maximum airspeed was generally limited by power available at lower altitudes and by the "never exceed" (Vne) airspeed from the Federal Aviation Agency Type Inspection Authorization (FAA TIA) (See Part I, Section 1, paragraph 1.1.g) at higher altitudes.

Vibration levels were exceptionally low and blade stall or rotor compressibility effects were not encountered.

While changing collective pitch control settings, the pilot had to monitor rotor speed and use the "beep" switch to maintain a desired rpm.

Autorotational descent performance was satisfactory and typical for a helicopter the size and weight of the OH-4A. The minimum rate of descent was 1495 feet per minute. This was obtained at 48.0 knots calibrated airspeed (KCAS) and the minimum power-off rotor speed of 374 rpm.

1.8 CONCLUSIONS

None.

1.9 RECOMMENDATIONS

None.

2.0 INTRODUCTION

Performance flight tests on the OH-4A 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 121 flights were conducted for 73:45 productive flight hours. The tests were accomplished during the calendar period of 13 March 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 obtained entirely at Bakersfield where the 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 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 guarantees and the desired performance as defined in "Military Characteristics, Light Observation Aircraft" (See Part I, Section 1, paragraph 1.1a) for hovering, maximum airspeed and endurance.

Summary performance was obtained using power available and fuel flow information based on the T63-A-5 engine referred characteristics as defined by model specification 580-E 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.

Using the power available as described above, it was determined that a T63-A-5 engine, which 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 the 580-E model specification. Such an engine will, in fact, only produce 199 SHP under this altitude, ambient temperature condition (See Section 3, Appendix II).

2.1 TAKEOFFS

2.1.1 OBJECTIVE

Takeoff tests were conducted to determine the performance of the OH-4A 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 airspeeds 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 range 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 to 10,000 feet. Gross weight was varied from 2250 to 2930 pounds at the mid center-of-gravity (C.G.) location (Station 102.0). All takeoff tests were conducted with a main rotor speed of 394 rpm (100 percent power turbine speed (N_2)). using takeoff power.

2.1.3 RESULTS

The results of the takeoff performance tests are graphically presented in Figures No. 1 through 17, Section 3, Appendix I.

2.1.4 ANALYSIS

a. Comparative Techniques

Two takeoff techniques were used during this evaluation. Each technique was found to have certain advantages and was preferred under different circumstances.

The first technique required that sufficient power be applied to maintain the helicopter light on its skids. This placed the engine as high as possible in the power range while maintaining ground contact with the helicopter. This method of power management reduced the time required to accelerate the engine to takeoff power. As takeoff power was applied and the helicopter left the ground, a pitch attitude was assumed such that the helicopter climbed and accelerated simultaneously and reached the desired airspeed while passing through a height of 50 feet above the ground. This technique is referred to as the "simultaneous climb accelerate from light on skids" technique.

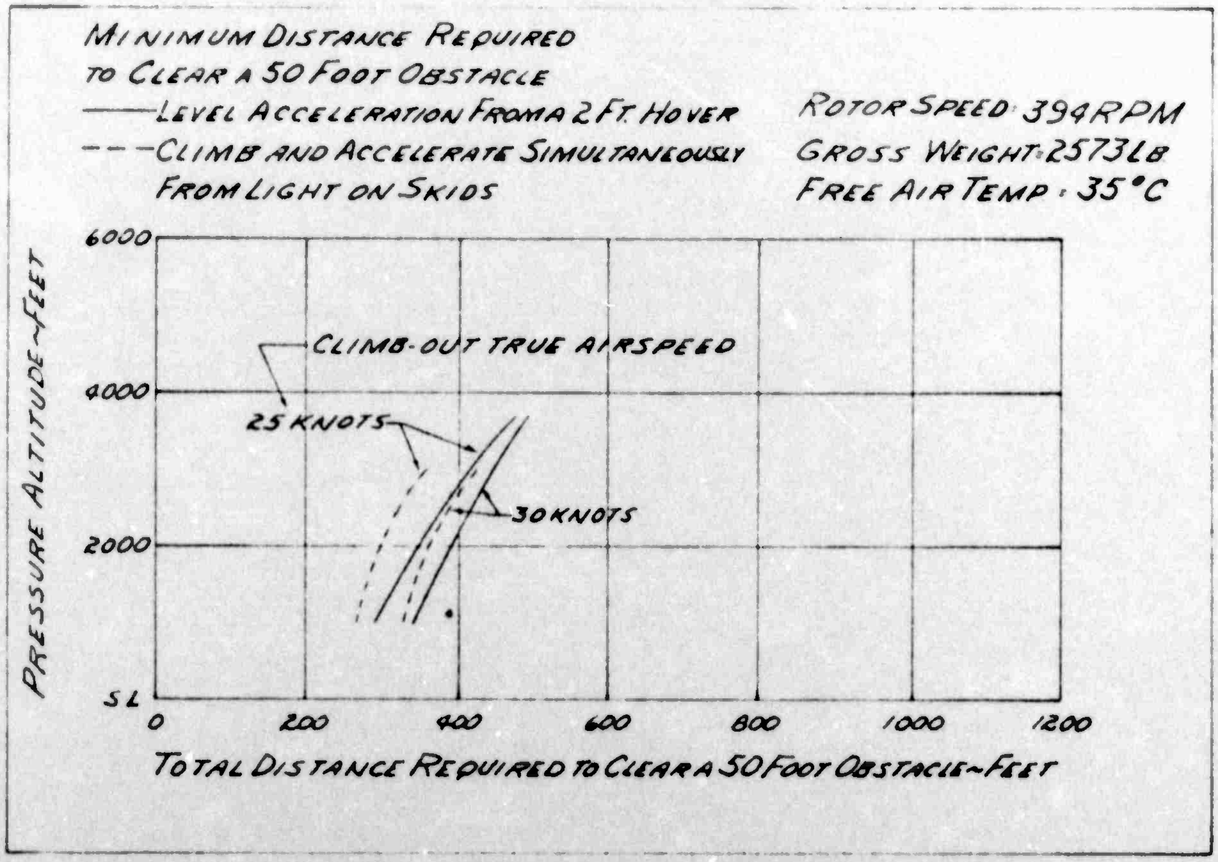
This technique produced shorter takeoff distances at conditions of higher values of excess power (equivalent hovering skid heights of 2-1/2 feet or more). In addition, the technique had several other inherent advantages. A flight path clear of intermediate obstacles was not required. Because of the terrain avoidance features of this technique, the pilot was able to monitor more closely the aircraft instruments, especially with regard to maintaining maximum engine power output and proper rotor speed.

One disadvantage of this technique was that considerable pilot proficiency was required to use it with precision. Attaining the desired airspeed at a 50 foot skid height was entirely dependent upon pilot judgment and skill in assuming the proper initial attitude at lift-off. An additional disadvantage was that the helicopter was within the "avoid" area of the height-velocity curve very early in the takeoff maneuver.

The second takeoff technique was initiated from a 2 foot hovering skid height. As power was applied, the helicopter was accelerated at a constant 2 foot skid height until approximately 3 knots before the desired climb-out airspeed was attained. The helicopter was then rotated to the proper attitude and allowed to climb out at the desired airspeed. When using this technique with a large amount of excess power, pronounced nose-down attitudes were required in order to maintain a constant skid height throughout the acceleration to climb-out airspeed. When there was very little excess power available, the helicopter had a tendency to lose height when passing through translation. If the helicopter were allowed to contact the ground, acceleration decreased and the takeoff distance increased.

Acceleration through translation was accomplished as rapidly as possible. Practice was required to achieve the maximum acceleration attainable for the various amounts of excess power available during the tests.

At large values of excess power available, equivalent to hovering skid heights of 2-1/2 feet or more, the minimum takeoff distances were obtained using the "simultaneous climb and accelerate from light on skids" technique. The following plot indicates that the "level acceleration from a 2 foot hover" technique may produce shorter takeoff distances under some conditions. At high values of excess power, however, the airspeed for minimum takeoff distance was lower for the "simultaneous climb and accelerate from light on skids" technique, and shorter overall distances were required.



Data were not collected for the "climb and accelerate" technique under conditions of low excess power (equivalent hovering skid height of approximately 3 feet or less) and low climb-out-airspeeds (less than 15 knots). Program termination dates and the available atmospheric conditions for testing precluded gathering these data. There was a minimum value of excess power below which this technique would have no longer been optimum or even practical. Extrapolation of the data in the preceding plot indicated that this point in excess power was at an equivalent hovering skid height of approximately 2-1/2 feet. At lower values of excess power than approximately 2-1/2 feet equivalent hovering skid height, ground effect was required in order to accelerate through translation to the desired climb-out airspeed, and translational lift was necessary before climb-out could be accomplished.

b. Flight Characteristics

The flight characteristics of the OH-4A during takeoff were, in general, satisfactory for achieving optimum takeoff performance. The static longitudinal stability and good controllability about all axes permitted proper attitude control. Control of aircraft attitude along with proper power management was vital to achieving consistent maximum takeoff performance. The importance of attitude control is emphasized because the shortest horizontal distances required to clear a 50 foot obstacle were achieved at airspeeds lower than that at which the standard aircraft's airspeed system became effective, i.e., less than 25 knots true airspeed (KTAS). The takeoff airspeed and the resulting performance were, therefore, entirely dependent upon pilot judgment and proficiency. It should be noted that the takeoff performance presented in this report can only be obtained by a proficient pilot.

Translation to forward flight was comparatively smooth, with no abrupt trim changes. At very low excess power, some settling was evident which occasionally resulted in ground contact. Following translation, the additional lift significantly increased the acceleration of the helicopter to climb-out airspeed. For this reason, it was important to pass through translation as rapidly as possible to obtain maximum performance.

c. Power Management

Power management during maximum performance takeoff in the OH-4A presented difficulties which were detrimental to the overall takeoff performance of the helicopter. An excessive amount of pilot attention was required to monitor engine operation during abrupt power changes.

To achieve maximum takeoff performance, it was important that the takeoff power limit be attained and maintained as rapidly and accurately as possible following initiation of the maneuver. Maintaining the maximum engine power output required that the pilot continuously make small adjustments to the collective control setting as the aerodynamic loading on the main rotor varied in different flight regimes. This was especially noticed while passing through translation. These corrections in collective pitch caused the rotor speed to vary significantly during a takeoff and from one takeoff to the next under the same start conditions of power and rotor speed.

It was the function of the power turbine (N_2) governor and the droop compensator cam to maintain a nearly constant rotor speed during small variations in power (collective control) settings. This function was not consistently performed, so that it was very difficult to maintain a constant rotor speed throughout the takeoff run. The variation in rotor speed with small power changes was not consistent from one takeoff to the next. The pilot could not effectively anticipate and correct rotor speed changes with the power turbine speed selector "beep" switch. In the course of the takeoff performance testing, an occasional inadvertent rotor overspeed at the takeoff power limit was encountered while attempting to maintain 394 (100 percent) revolutions per minute (rpm). (See Section 3, Appendix II, for a more complete description of fuel control operation.)

During takeoff under conditions of large amounts of available excess power, the gas producer must accelerate through a considerable range. The response characteristics of the T63-A-5 engine installed in the OH-4A were found to be unsatisfactory. The average acceleration time from flight idle (63 percent N_1) to 99.5 percent gas producer speed (N_1) was approximately 5 seconds, which is within acceptable limits. However, the initial acceleration after a rapid throttle application (throttle jam) was slow, with the acceleration rate increasing very rapidly during the last 2 seconds. This acceleration characteristic resulted in a large increase in torque during the last second in the acceleration cycle. This was difficult for the pilot to anticipate and to apply the anti-torque control necessary to counteract the resulting yawing motion of the helicopter.

2.2 HOVER

2.2.1 OBJECTIVE

Hovering tests were conducted to determine hovering performance of the OH-4A helicopter. Data from these tests were used to check compliance with the Military Characteristics and to 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 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 center-of-gravity (C.G.) location (Station 102.0) and rotor speed of 374, 382 and 394 rpm. Testing was accomplished at density altitudes from sea level to 10,300 feet.

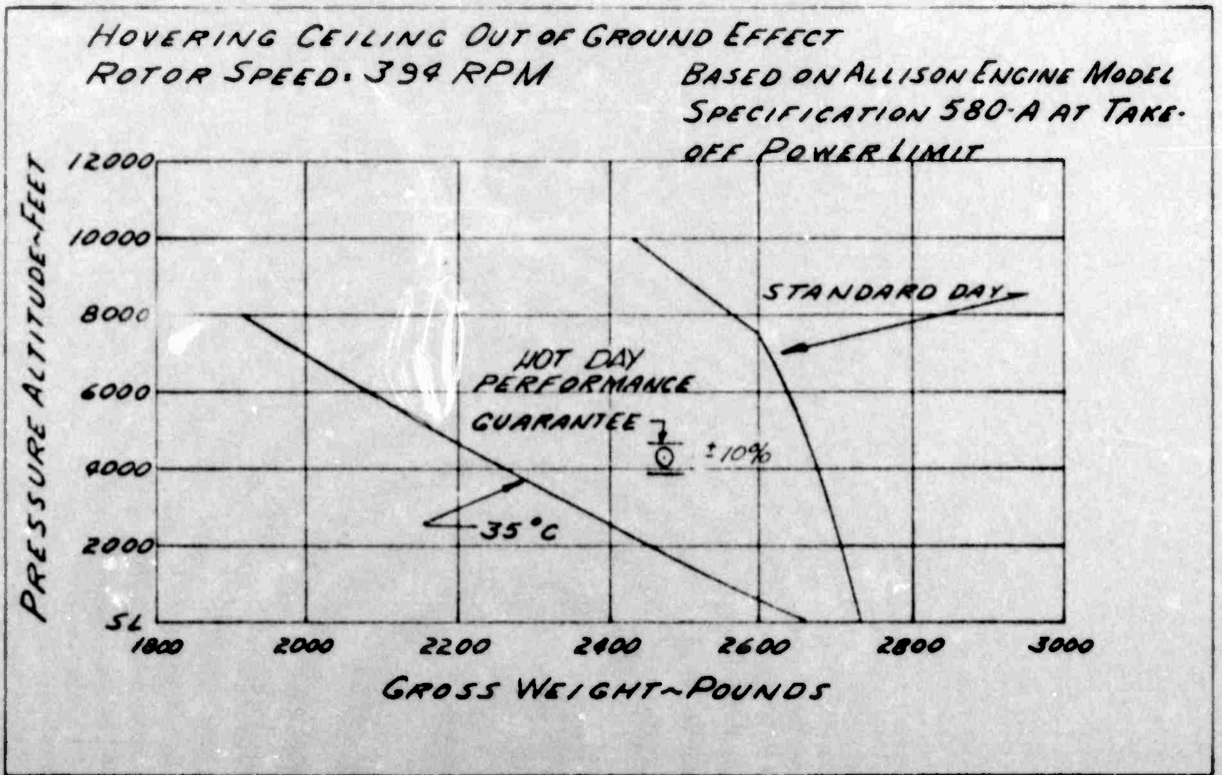
2.2.3 RESULTS

The results of the hovering performance tests are graphically presented in Figures No. 18 through 28, Section 3, Appendix I.

2.2.4 ANALYSIS

a. Guarantee

The OH-4A did not meet the guaranteed hovering performance as shown on the following plot:



The hovering performance guarantee stated that the OGE hovering ceiling at normal gross weight (2573 pounds) with an ambient temperature of 35 degrees centigrade (C) (95 degrees Fahrenheit (F)) would be 4300 feet pressure altitude, ± 10 percent. Power available was defined by Model Specification 580-A, with installation losses as measured during this evaluation.

The maximum gross weight at which the OH-4A could hover OGE at 4300 feet pressure altitude on a 35 degree C day was 2234 pounds. The actual hovering ceiling under the conditions of the hovering performance guarantee was at a pressure altitude of 825 feet.

The Military Characteristics, Light Observation Aircraft (See Part I, Section 1, paragraph 1.1a) required that the normal gross weight 2573 pounds) OGE hovering ceiling on a 35 degree C day be 6000 feet pressure altitude. The maximum gross weight at which the OH-4A could hover OGE at 6000 feet on a 35 degree C day was 2085 pounds.

The Military Characteristics also required an OGE hovering capability at overload gross weight (2900 pounds) at sea level on a standard day. This requirement was not met using power available (250 SHP) as defined by Model Specification 580-A. The maximum gross weight at which the OH-4A could hover OGE on a standard day at sea level was 2730 pounds. This was 170 pounds (approximately 5.9 percent) less than the requirement.

The summary hovering performance presented in Figure No. 20, Section 3, Appendix I, to check compliance with the performance guarantee, was significantly different from the takeoff power summary hovering performance presented in Figure No. 18, Section 3, Appendix I. The difference was due to the installed power available as defined by the two Model Specifications, 580-A and 580-E. Below critical altitude, the difference was due to the fact that the engine output torque was limited to 218 pound-feet for the 580-A, and to 240 pound-feet for the 580-E. Above critical altitude, the difference was due to the difference in the power available between the two Model Specifications at the same limit turbine outlet temperature (Tt_5). This difference was larger for the higher ambient temperatures.

b. Performance

The hovering performance summaries presented in Figures No. 18 and 19, Section 3, Appendix I, were based on the power available as defined by Model Specification 580-E, with installation losses as measured during this evaluation.

The OGE hovering performance of the OH-4A under standard day sea level conditions was adequate, however, performance deteriorated very rapidly with increasing ambient temperature as well as altitude. The standard day OGE hovering ceiling at the overload gross weight of 2900 pounds was 2200 feet, but on a 35 degree C day, the OH-4A would not hover OGE at sea level at the normal gross weight of 2573 pounds. The maximum gross weight at which the OH-4A could hover OGE at 6000 feet pressure altitude on a 35 degree C day was 1984 pounds.

At normal gross weight (2573 pounds), the 2 foot skid height hovering ceiling was 4150 feet pressure altitude on a 35 degree C day and 12,150 feet on a standard day. At overload gross weight (2900 pounds) the 2 foot skid height hovering ceiling was 1210 feet pressure altitude on a 35 degree C day and 8200 feet on a standard day.

The rapid deterioration of hovering performance with increasing ambient temperature was largely due to the rapid decrease in power available with temperature rise. This power deterioration was considered typical for a helicopter powered by a gas turbine engine when operating above critical altitude.

Comparison of the hovering performance deterioration of the OH-4A and the OH-13H illustrates the difference in power loss with increased temperature for turbine and reciprocating engines. At 8000 feet pressure altitude, above the critical altitude for both helicopters, the maximum gross weight for OGE hovering of the OH-13H was 2532 pounds on a standard day and 2340 pounds on a 35 degree C day (See Section 1, paragraph 1.1p). The maximum gross weight for OGE hovering for the OH-4A was 2495 pounds on a standard day and 1810 pounds on a 35 degree C day at 8000 feet pressure altitude. The OH-13H OGE hovering capability was decreased by 192 pounds or approximately 7.6 percent. The OH-4A OGE hovering capability was decreased by 685 pounds or approximately 27.5 percent.

Rotor speed had a negligible effect upon the OH-4A hovering performance, but rotor speed stability while hovering in gusty winds was poor. The changes in power delivered to the main rotor because of changing aerodynamic loading and control power requirement caused the rotor speed to vary indiscriminately through a range of up to 5 rpm. This variation, however, did not adversely affect skid height or altitude control.

2.3 CLIMBS

2.3.1 OBJECTIVE

Climb tests were conducted to determine the performance during climbing flight and the service ceiling.

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 the same gross weights. During the climb, power was maintained at limit engine torque until an altitude was reached at which the limiting turbine outlet temperature was obtained (critical altitude). As the climb continued above this altitude, power was adjusted to maintain the limit turbine outlet temperature.

The continuous climbs were conducted at gross weights of 2573 pounds (normal) and 2900 pounds (overload) at a mid C.G. location (Station 102.0) and a rotor speed of 394 rpm.

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

2.3.3 RESULTS

The results of the climb performance tests are presented graphically in Figures No. 29 through 33, Section 3, Appendix I.

2.3.4 ANALYSIS

The climb performance of the OH-4A was satisfactory. With a start gross weight of 2573 pounds on a standard day at sea level, the service ceiling of the OH-4A was 20,150 feet. At maximum continuous power, the rate of climb at sea level was 1100 feet per minute, increasing to a maximum of 1150 feet per minute at the critical altitude for maximum continuous power (3150 feet). At the same conditions, but using takeoff power, the sea level rate of climb was 1550 feet per minute, increasing to 1575 feet per minute at the critical altitude for takeoff power (2250 feet).

With a climb start gross weight of 2900 pounds at sea level on a standard day, the service ceiling of the OH-4A was 15,600 feet. At maximum continuous power, the sea level rate of climb was 810 feet per minute, increasing to a maximum of 850 feet per minute at the critical altitude for maximum continuous power (3150 feet). At the same conditions, but using takeoff power, the sea level rate of climb was 1250 feet per minute, increasing to a maximum of 1270 feet per minute at the critical altitude for takeoff power (2250 feet).

The airspeed flown to obtain the maximum rate of climb was based upon level flight performance data. Due to time limitations, a complete analysis of level flight data was not possible prior to completing the climb tests, and the optimum climb-speed schedule was not flown in every case. Figures No. 29 and 30, Section 3, Appendix I, show the calibrated airspeed actually flown along with optimum climb speed schedule. At lower altitudes and gross weights, maintaining the airspeed for maximum rate of climb was not critical due to the relatively constant value of shaft horsepower (SHP) required in level flight at true airspeeds near the airspeed for maximum rate of climb. Near the service ceilings, the relationship between SHP required in level flight and true airspeed was more critical, and the actual service ceilings may be slightly higher than those shown in Figures No. 29 and 30, Section 3, Appendix I.

Maintaining the selected airspeed during a climb presented no particular problems once the helicopter was stabilized in the climb. However, at light weight and high power, it was rather difficult to establish a stabilized airspeed at low altitudes. Due to the initial high rate of climb, considerable altitude was gained before the helicopter was completely stabilized. It was found that the best method for stabilizing at the desired airspeed and power setting was to increase power at a rate that allowed the airspeed to remain stabilized.

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-4A helicopter. Data from these tests were used to check compliance with the Military Characteristics and to determine if the contractual guarantees were met. Tests were also conducted to determine the effect on performance of the XM-7 and XM-8 weapons system.

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 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 2200, 2500 and 2900 pounds were used at a mid center-of-gravity (C.G.) location (Station 102.0). Rotor speeds of 374, 382 and 394 rpm were flown. Two additional tests were conducted at a density altitude of 5000 feet, a gross weight of approximately 2400 pounds, with a rotor speed of 394 rpm. One of these tests was at a forward C.G. location (Station 99.3) and the other at an aft C.G. location (Station 106.3).

Tests in the armed configuration were conducted at density altitudes of approximately 2000 and 6000 feet, a gross weight of approximately 2400 pounds, a mid C.G. location and with a rotor speed of 394 rpm.

2.4.3 RESULTS

The results of the level flight performance tests are presented graphically in Figures No. 34 through 51, Section 3, Appendix I.

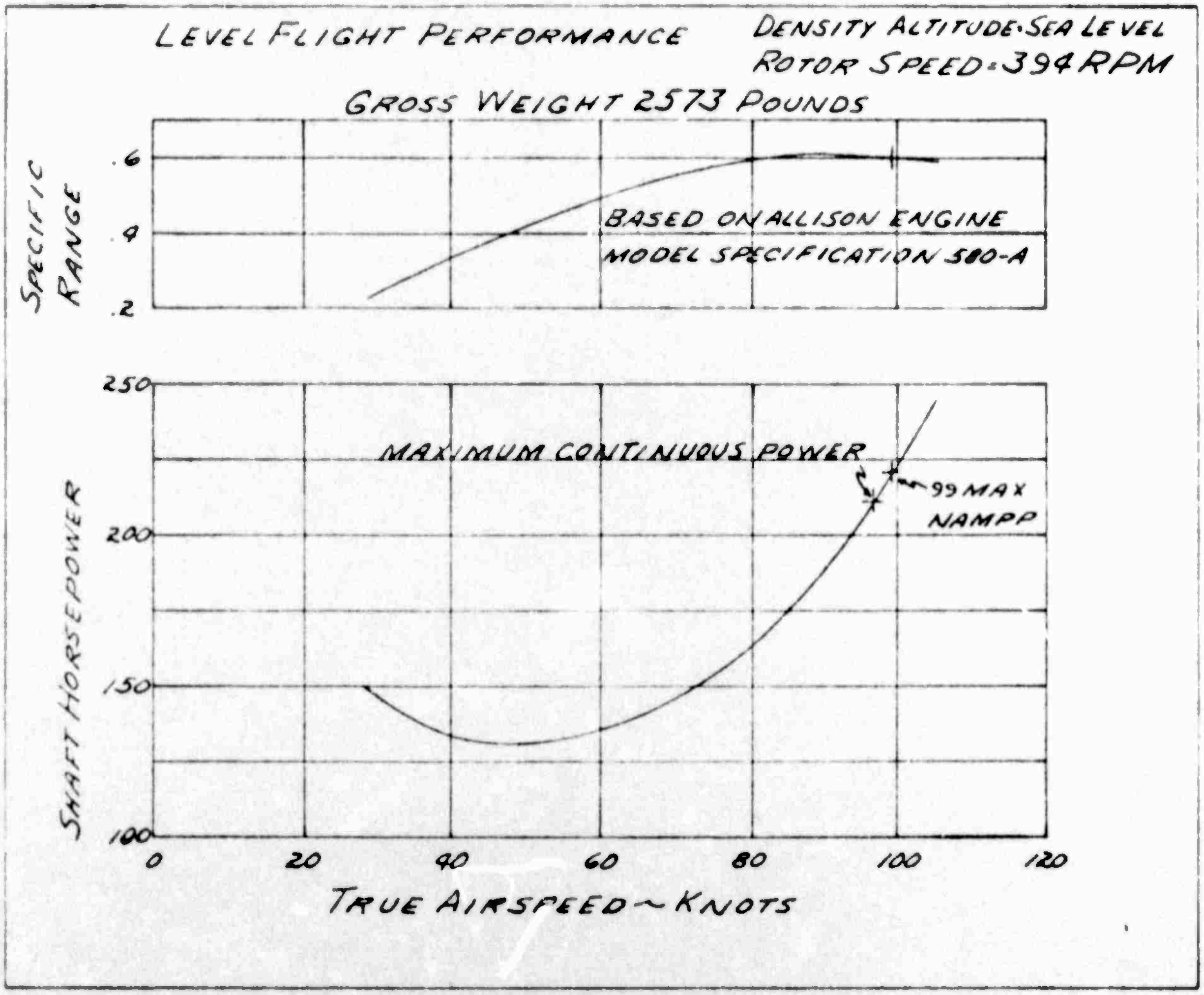
2.4.4 ANALYSIS

a. Guarantee

The OH-4A met the maximum airspeed guarantee. The OH-4A Model Specification maximum airspeed guarantee was 100 knots ± 10 percent true airspeed for the conditions of sea level on a standard day at normal gross weight (2573 pounds) and maximum continuous power as defined by Model Specification 580-A. The relationship of SHP and specific range to true airspeed at these conditions is presented in the plot on Page 17.

The maximum true airspeed was found to be 96.5 knots. This exceeded the minimum guarantee (100 knots - 10 percent) by 6.5 knots true airspeed (KTAS) (approximately 7.2 percent).

The Military Characteristics, Light Observation Aircraft required a cruise speed of 110 KTAS at the same conditions as above. The OH-4A failed to meet this requirement by a margin of 13.5 knots (approximately 12.3 percent).



The OH-4A failed to meet the endurance mission performance guarantee. The conditions for the endurance mission guarantee were stated in the OH-4A Model Specification as presented in the table on Page 18.

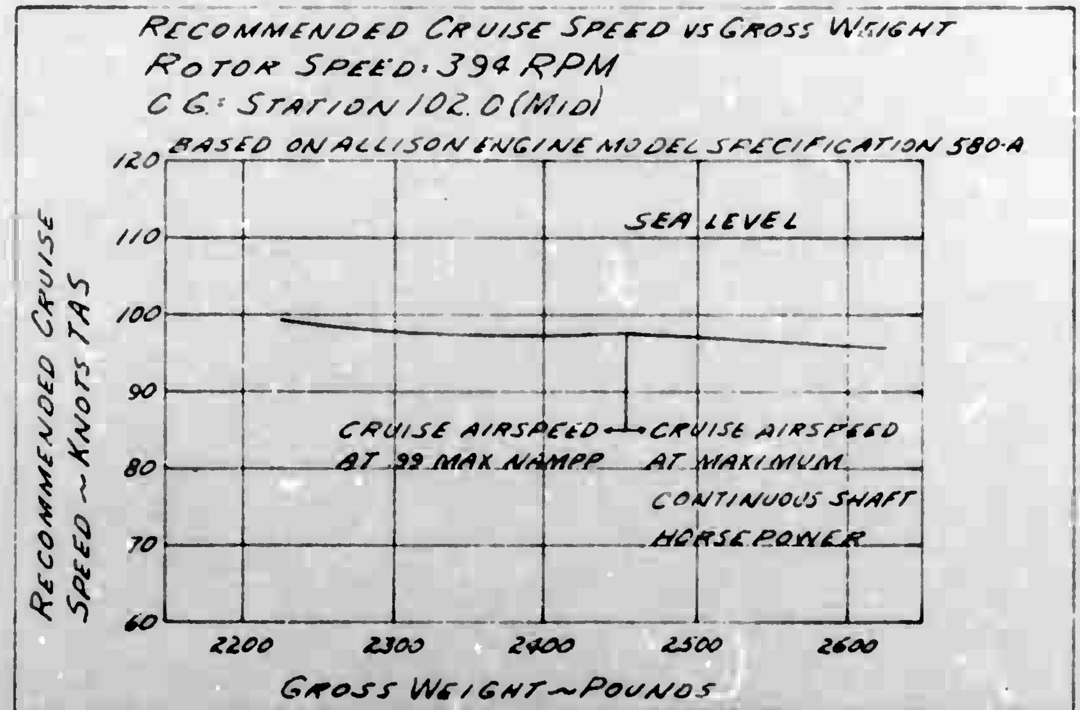
The guaranteed endurance at the conditions stated in Table I, on Page 18, was 3.0 hours ± 10 percent. The actual endurance of the OH-4A at the above conditions was 2.33 hours. This failed to meet the minimum guaranteed endurance of 2.7 hours (3.0 hours - 10 percent) by a margin of 0.37 hours (approximately 13.6 percent).

TABLE NO. I

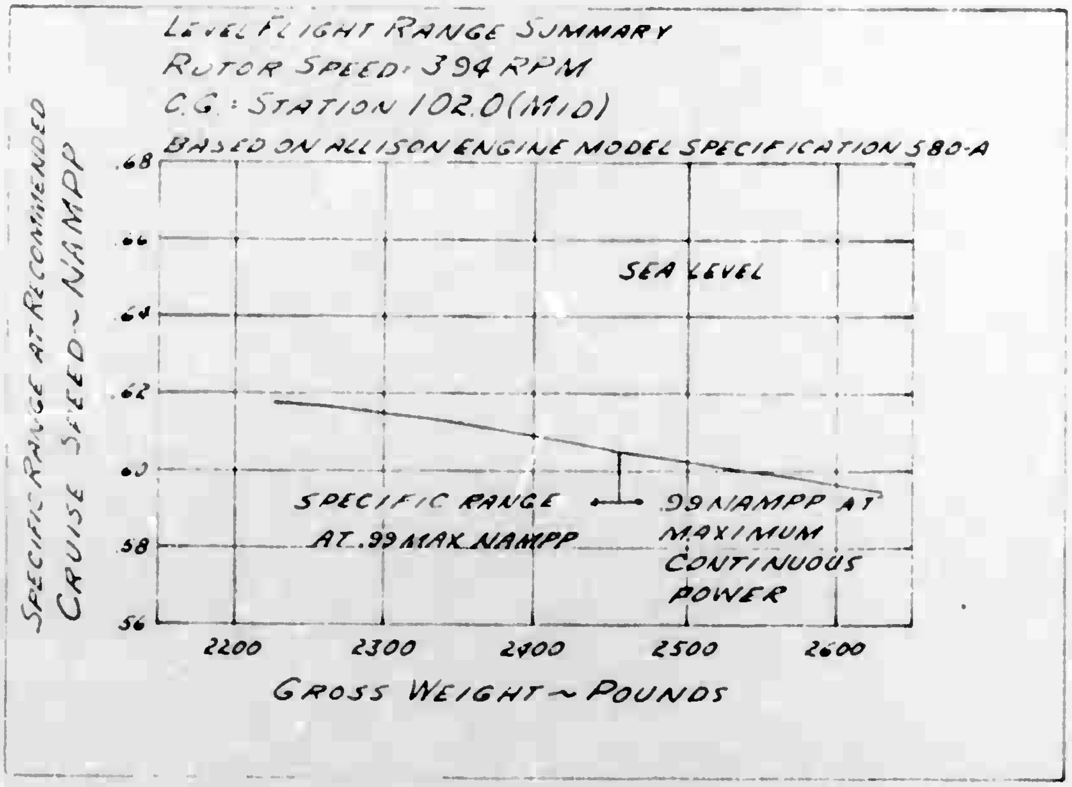
Basic weight (full oil and trapped fuel)	1596
Pilot	200
Observer and cargo	400
Fuel (including 1 lb unusable fuel)	377
Engine start gross weight	2573 lb
Warm up and takeoff: 3 minutes at maximum continuous power at sea level	
Cruise 96 kt at sea level	
No fuel reserve	

The empty weight of the OH-4A was considerably higher than originally intended. Because of this, the amount of fuel that could be carried had to be reduced to stay within the OH-4A Model Specification normal gross weight of 2573 pounds (2450 pounds + 5 percent). This contributed to a significant decrease in the range and endurance capability of the helicopter.

The airspeed specified for the endurance mission, 96 KTAS, did not represent the airspeed for maximum range performance. The airspeed for maximum range performance, the recommended cruise speed, is shown in the following plot:



At gross weights greater than 2455 pounds, the cruise speed was limited by maximum continuous power. A specific example of this condition may be seen in the plot on page 17. At gross weights lower than 2455 pounds, the recommended cruise speed was that airspeed at which specific range was 99 percent of its maximum value for a specific gross weight. The values of specific range at the recommended cruise speeds are shown in the following plot:



The Military Characteristics, Light Observation Aircraft (See Part I, Section 1, paragraph 1.1a) listed a requirement for 3 hours endurance with a 400 pound payload. The conditions for this requirement were the same as those for the endurance mission performance guarantee in Table I with the exception of airspeed. The airspeed for minimum power required was used rather than the cruise speed of 96 KTAS. At the airspeed for minimum power required, the endurance of the OH-4A was 3.18 hours, exceeding the requirement by 0.18 hours (approximately 6.0 percent).

A second endurance requirement of the Military Characteristics, Light Observation Aircraft was that at 85 percent cruise power the endurance at the conditions listed in Table I, with the exception of cruise speed, should be 3 hours. The cruise speed for this requirement was the airspeed obtained at 85 percent of the power required at the recommended cruise speed. The true airspeed at 85 percent cruise power ranged from 86.7 knots at the start of the mission to 89.7 knots with nearly all the fuel used. The endurance at 85 percent cruise power was 2.55 hours. This was 0.45 hours (approximately 15 percent) below the requirement.

b. Performance

The maximum airspeed of the OH-4A was limited either by SHP available or by the "never exceed" airspeed (Vne) as defined by the OH-4A Federal Aviation Agency Type Inspection Authorization (FAA TIA) (See Part I, Section 1, paragraph 1.1g). At density altitudes from sea level to approximately 5000 feet, the limit was usually SHP available. Above that altitude, the limit was usually Vne. The limit encountered depended upon gross weight and ambient temperature as well as density altitude. At a given pressure altitude and gross weight, an increase in ambient temperature would increase the density altitude, and thereby decrease the TIA Vne limit. However, above critical altitude, an increase in temperature would also reduce the power available. The effect of increased temperature upon power available, at a pressure altitude, is greater than the effect upon the TIA Vne. This meant that, at higher ambient temperatures, the pressure altitude at which the TIA Vne limit was reached first was higer than that on a colder day.

The level flight performance tests conducted with the XM-7 armament system installed in the stowed position showed that the equivalent flat plate area of this system was approximately 2.46 square feet. With this increase in drag, the XM-7 installation would reduce the sea level standard day maximum airspeed at normal gross weight (2573) using takeoff power from 111.5 KTAS to 105 KTAS. At maximum continuous power, under the same conditions, the maximum true airspeed would be reduced from 102 knots to 96 knots.

The level flight performance tests conducted with the XM-8 armament system installed in the stowed position showed that the equivalent flat plate area of this system was approximately 1.57 square feet. On a sea level standard day at design gross weight (2573 pounds), the XM-8 armament system reduced the maximum true airspeed at takeoff power from 111.5 knots to 107.5 knots. At maximum continuous power, under the same conditions, the maximum true airspeed would be reduced from 102 knots to 98 knots.

Two level flight tests were conducted at similar conditions of gross weight and density altitude. In one case the helicopter C.G. was at fuselage Station 99.3. In the second case the helicopter C.G. was at fuselage Station 106.3. It was found that there was negligible difference in the SHP required in level flight at these extremes of C.G. location.

Figure No. 35, Section 3, Appendix I, summarizes the range performance of the OH-4A. The airspeed for maximum range performance for the clean configuration at 394 rpm was a constant 101 KTAS for all gross weights and altitudes tested. The clean configuration includes the F.M. homing antennae, but did not include the XM-7 or XM-8 armament systems. The airspeed for maximum range was not the recommended cruise speed in every case. In some cases the Vne or the airspeed at maximum continuous power limited airspeed to a value lower than that for maximum range performance.

The F.M. homing antennae reduced the range of the OH-4A approximately 1 percent, and reduced the airspeed for maximum range from 103 KTAS to 101 KTAS.

The XM-7 armament system reduced the range approximately 7 percent and reduced the airspeed for maximum range from 101 KTAS to 94 KTAS.

The XM-8 armament system reduced the range approximately 4.5 percent and reduced the airspeed for maximum range from 101 KTAS to 95 KTAS.

Figure No. 68, Section 3, Appendix I, shows the vibration levels for the OH-4A at one gross weight and altitude. Vibration levels were calculated for the fundamental rotor frequency and at the second and fourth harmonics. The one and two-per-rotor revolution vibration levels were less than .07 g's. The four-per-rotor revolution vibration increased at both high and low speeds. Maximum lateral vibration was approximately 0.2 g's at 104.5 KTAS. Maximum vertical vibration was approximately 0.15 g's at the same airspeed. Qualitatively, the overall vibration characteristics of the OH-4A were exceptionally good.

During this evaluation, the OH-4A did not exhibit any tendency to enter blade stall. In the envelope flown during the stability and control portion of this evaluation, there was no tendency toward pitch-up or loss of lateral controllability, the usual indications of incipient blade stall (See Part I). No compressibility effects on the rotor system were detected in the level flight power required tests. The OH-4A, therefore, could effectively use an increase in installed power. Level flight maximum speeds would be increased under conditions where airspeed was limited by SHP available.

The droop characteristics of the OH-4A were unsatisfactory. During power changes the selected rotor speed changed as much as ± 7 rpm.

It is the combined function of the droop compensator cam, the power turbine (N_2) governor and the gas producer speed (N_1) fuel control to schedule fuel flow to the engine. This fuel flow should meet the torque output demanded by the collective pitch control setting while maintaining a nearly constant engine output speed as selected by the pilot.

In practice, the pilot had to continually use the power turbine speed selector "beep" switch when changing collective control setting. When increasing airspeed in level flight from the airspeed for minimum power required to maximum airspeed, the rotor speed would decrease approximately 3 rpm. When transitioning from a climb at take-off power to level flight, the rotor speed would increase approximately 4 rpm. When initiating a partial power descent from level flight at cruise airspeeds, as in approach to a landing, the rotor speed would increase approximately 7 rpm.

The shortcoming in the governing of rotor speed was particularly significant when the helicopter was being controlled from the copilot position. There was no "beep" switch incorporated in the copilot collective control. The copilot had to release one flight control to activate the pilot's "beep" switch, or the pilot had to activate the switch.

Operation of the "beep" switch was satisfactory in the normal operating rpm range. However, a "dead spot" existed at both extremes of the "beep" range (minimum rpm and maximum rpm). This "dead spot" made very small rotor speed adjustments very difficult when operating in these areas.

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 the allowable ranges. During the descent, time and altitude were recorded so that rate of descent cc be determined.

A series of stabilized descents was conducted at a rotor speed of 394 rpm and at various airspeeds between 35 to 65 knots calibrated airspeed (KTAS). 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 374 to 422 rpm.

The autorotational descents were conducted at density altitudes from approximately 5000 to 10,000 feet. Various gross weights from 2180 to 2900 pounds were flown at a mid center-of-gravity (C.G.) location.

2.5.3 RESULTS

The results of the autorotational descent performance tests are presented graphically in Figures No. 52 through 54, Section 3, Appendix I.

2.5.4 ANALYSIS

The autorotational descent characteristics of the OH-4A were satisfactory and typical for a helicopter of its size and weight. The airspeed for minimum rate of descent was 48.0 KCAS. At this airspeed, the rate of descent varied from 1680 feet per minute at a rotor speed of 422 rpm to 1495 feet per minute at 374 rpm. The minimum angle of descent at a density altitude of 5000 feet and rotor speed of 394 rpm was achieved at 72 knots true airspeed (KTAS). Gross weight and density altitude had little effect upon rate of descent, but a lower rotor speed produced a lower rate of descent at a constant airspeed.

Above a density altitude of approximately 13,000 feet, it was not possible to achieve true autorotational flight with the engine operating. With the twist grip in the flight idle position and the collective pitch control full down, the engine rpm was too high to allow a complete engine-rotor disengagement ("needles split").

Power recoveries from autorotations were impractical and unsafe if initiated close to the ground (during or after the flare) when operating at density altitudes in excess of 3000 feet. This maneuver could be safely accomplished if throttle application were initiated prior to the flare.

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 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 30 to 105 knots calibrated airspeed (KCAS) using approximately 10 knot increments were flown. These tests were conducted at a density altitude of 1260 feet, a gross weight of 2550 pounds, a mid center-of-gravity (C.G.) location (Station 102.0) and a rotor speed of 394 rpm.

2.6.3 RESULTS

The results of the airspeed calibration are presented graphically in Figure No. 67, Section 3, Appendix I.

2.6.4 ANALYSIS

The calibration of the position error of the standard aircraft airspeed system showed good agreement with that presented by the manufacturer. The standard system had no position error at approximately 50 knots indicated airspeed (KIAS). The position error varied linearly from approximately +1-1/2 knots at 28 KIAS to -5 knots at 110 KIAS.

SECTION 3 - APPENDICES

APPENDIX I - TEST DATA

FIGURE NO 1
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION

TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS.
GROSS WEIGHT: 2573 LB
ROTOR SPEED: 394 RPM
– – – OUT OF GROUND EFFECT HOVERING CEILING

NOTE:
DERIVED FROM FIGURES 3, 25 AND 55

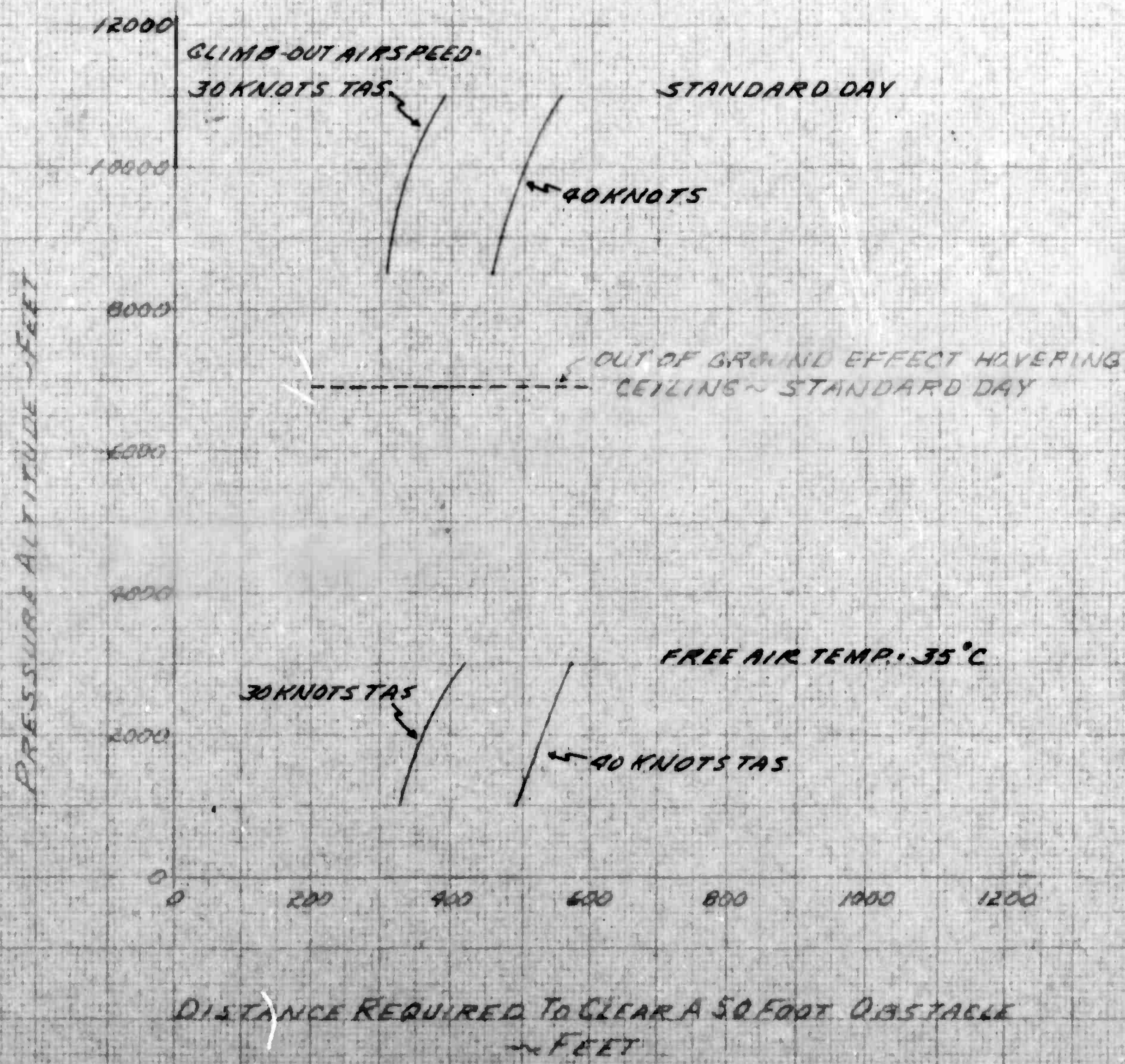


FIGURE NO. 2

SUMMARY TAKE-OFF DISTANCE REQUIRED

TO CLEAR A 50' OBSTACLE

OH-4A USA S/N 62-4202

CLEAN CONFIGURATION

TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS

GROSS WEIGHT = 2900 LB

ROTOR SPEED = 394 RPM

– – – OUT OF GROUND EFFECT HOVERING CEILING

NOTE:

DERIVED FROM FIGURES 3, 25 AND 55

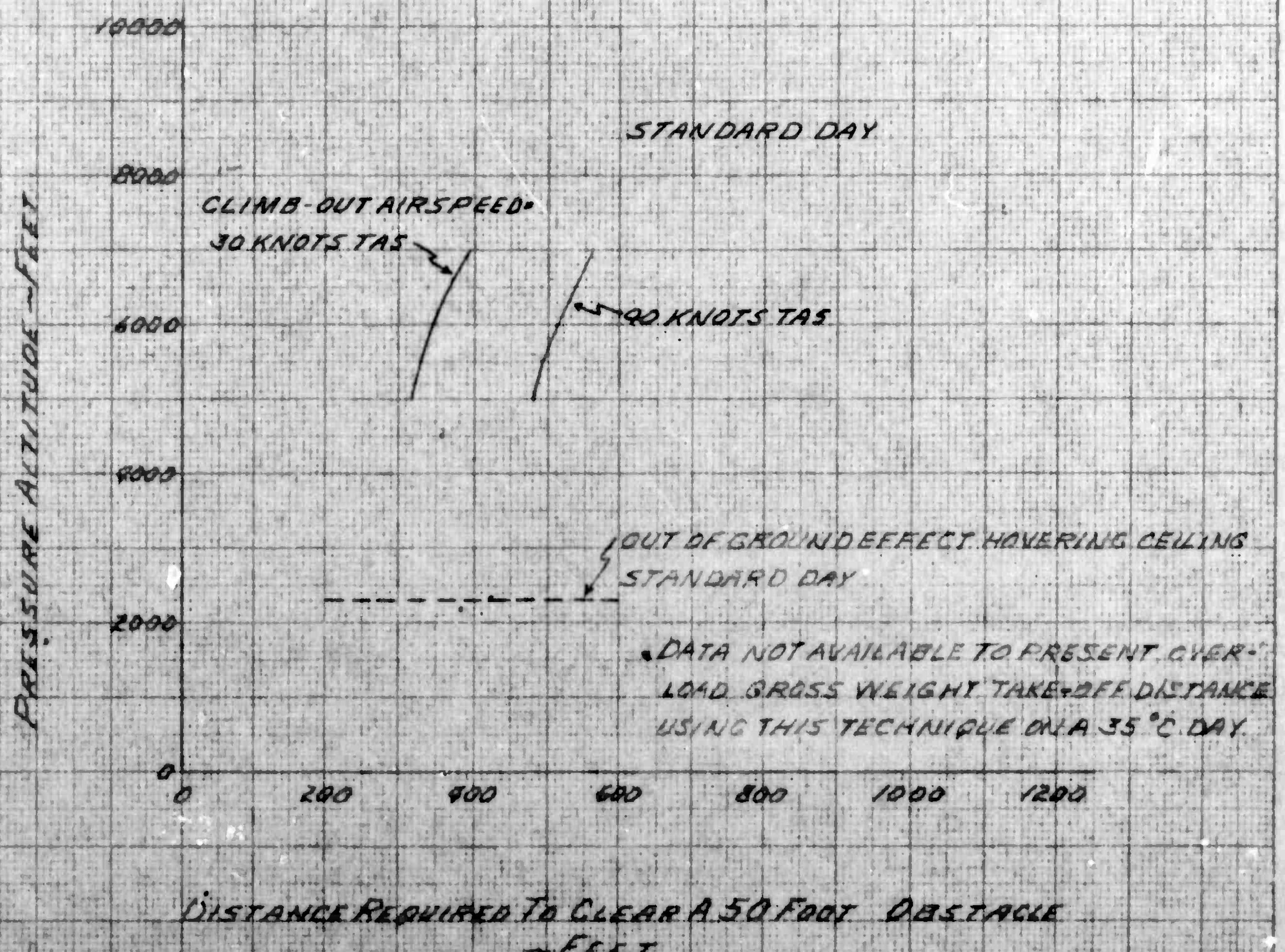


FIGURE NO. 3

VARIATION OF TAKE-OFF DISTANCE

WITH CLIMB-OUT AIRSPEED AND ΔC_p

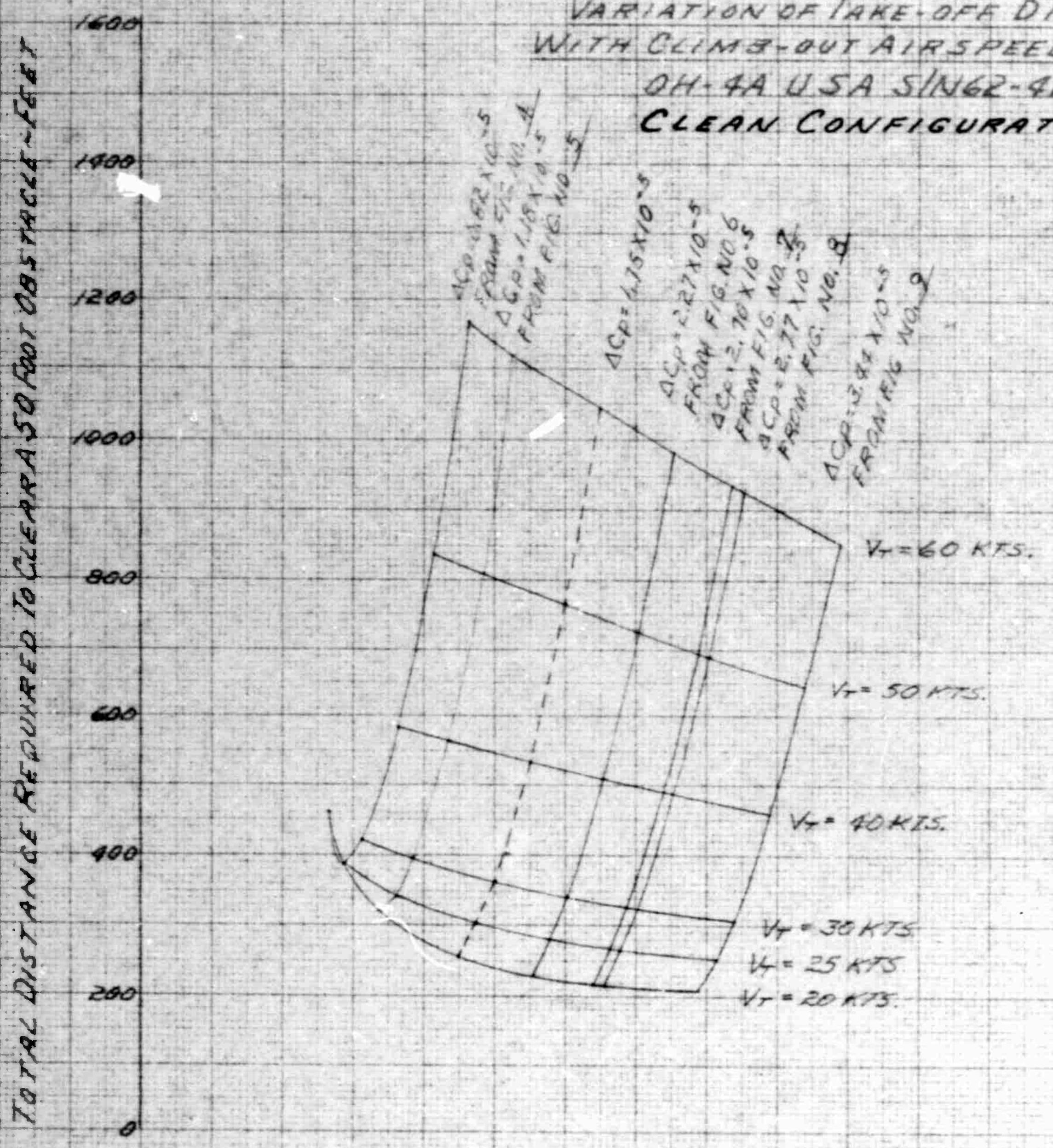
OH-4A USA S/N62-4202

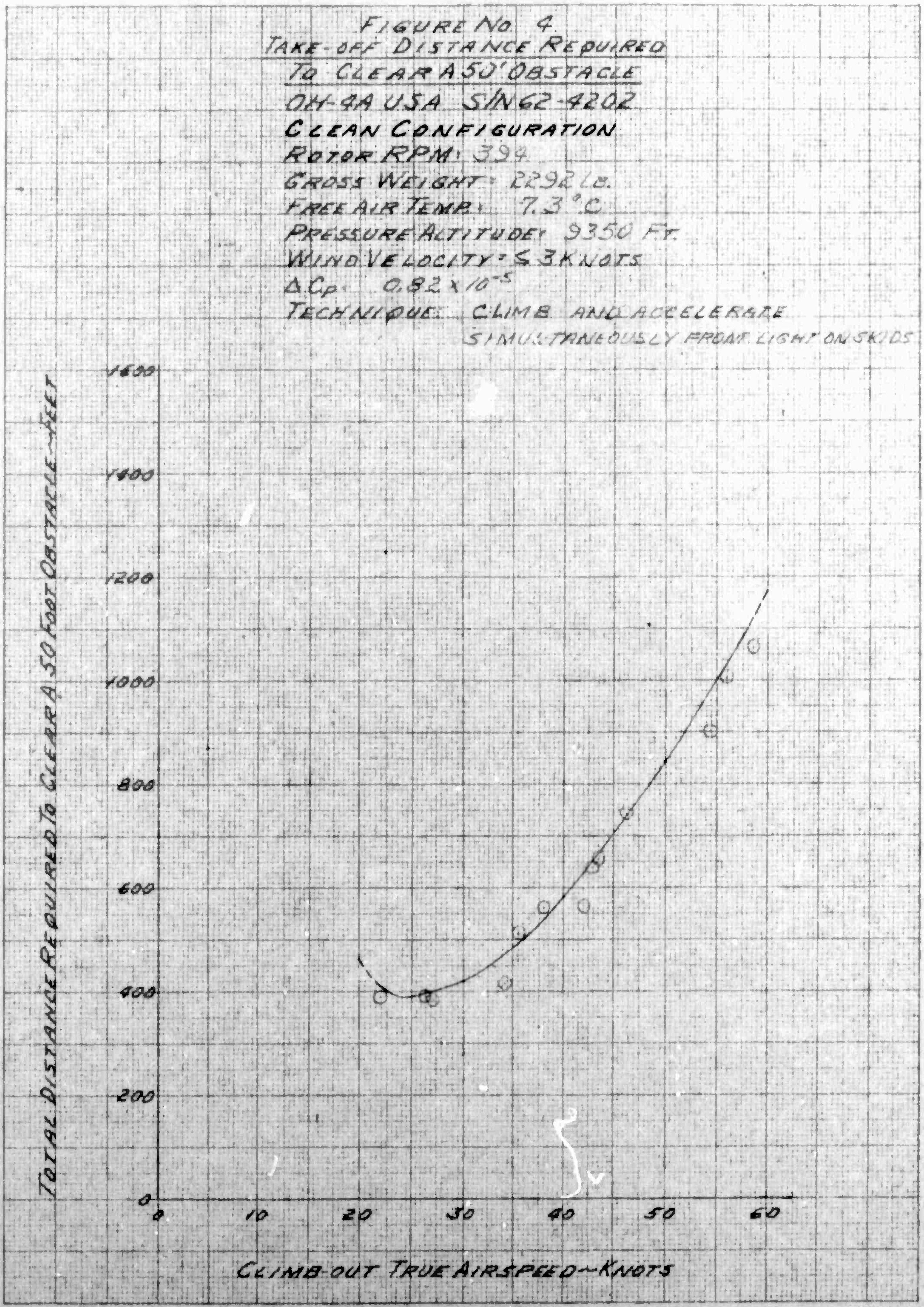
CLEAN CONFIGURATION ROTOR SPEED: 394 RPM

TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS

NOTES:

1. ΔC_p: C_p AT TAKE-OFF − C_p TO HOVER AT 2 FEET
2. C_p AT TAKE-OFF: BASED ON POWER AVAILABLE UNDER TAKE-OFF ATMOSPHERIC CONDITIONS
3. C_p TO HOVER AT 2 FEET: BASED ON POWER REQUIRED TO HOVER AT 2 FEET UNDER TAKE-OFF ATMOSPHERIC CONDITIONS.





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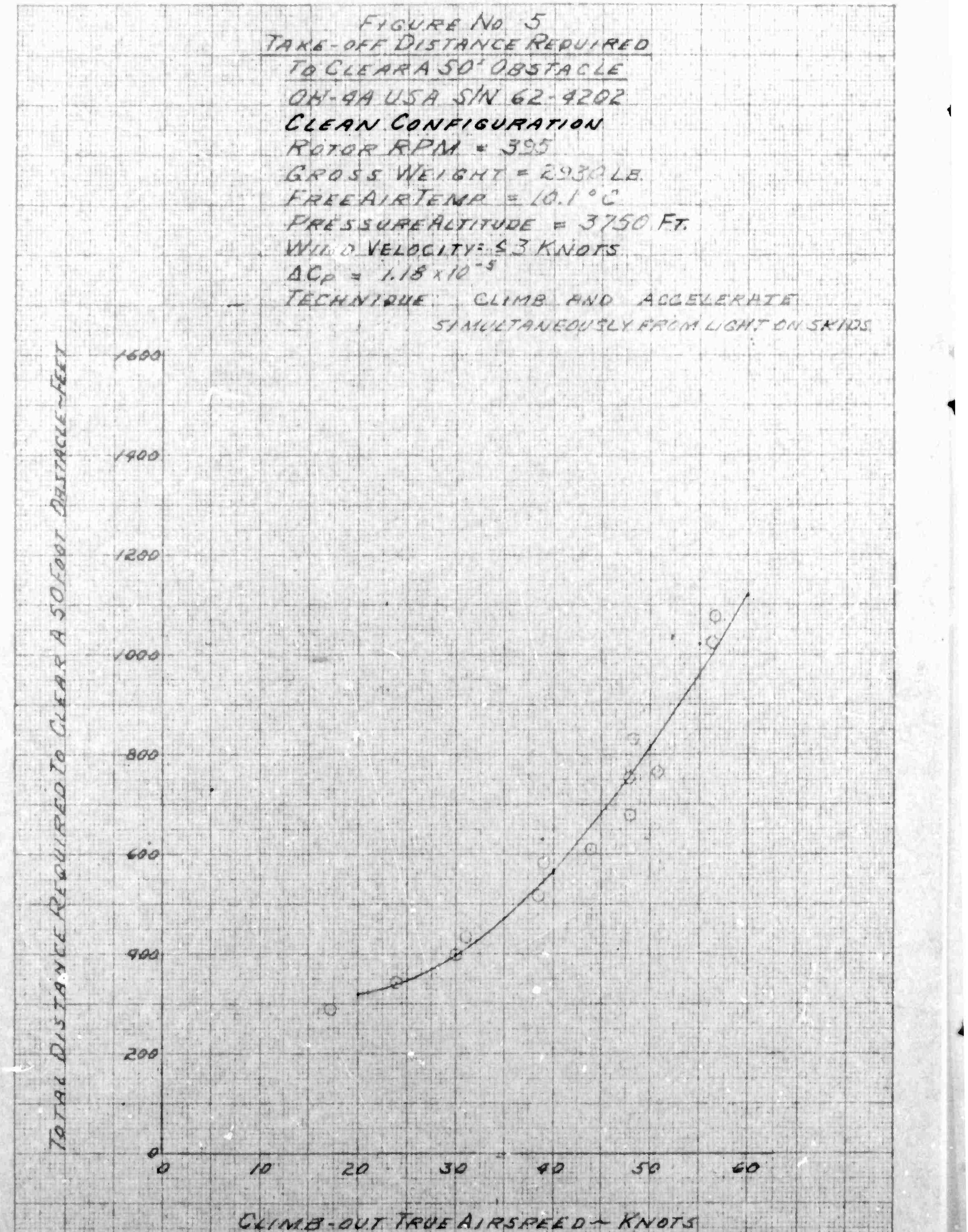

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FIGURE NO. 6
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE

OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 394
GROSS WEIGHT = 2526 LB
FREE AIR TEMP = 21.8 °C
PRESSURE ALTITUDE = 3970 FT.
WIND VELOCITY = ≤ 3 KNOTS
$\Delta C_p = 2.27 \times 10^{-5}$
TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS

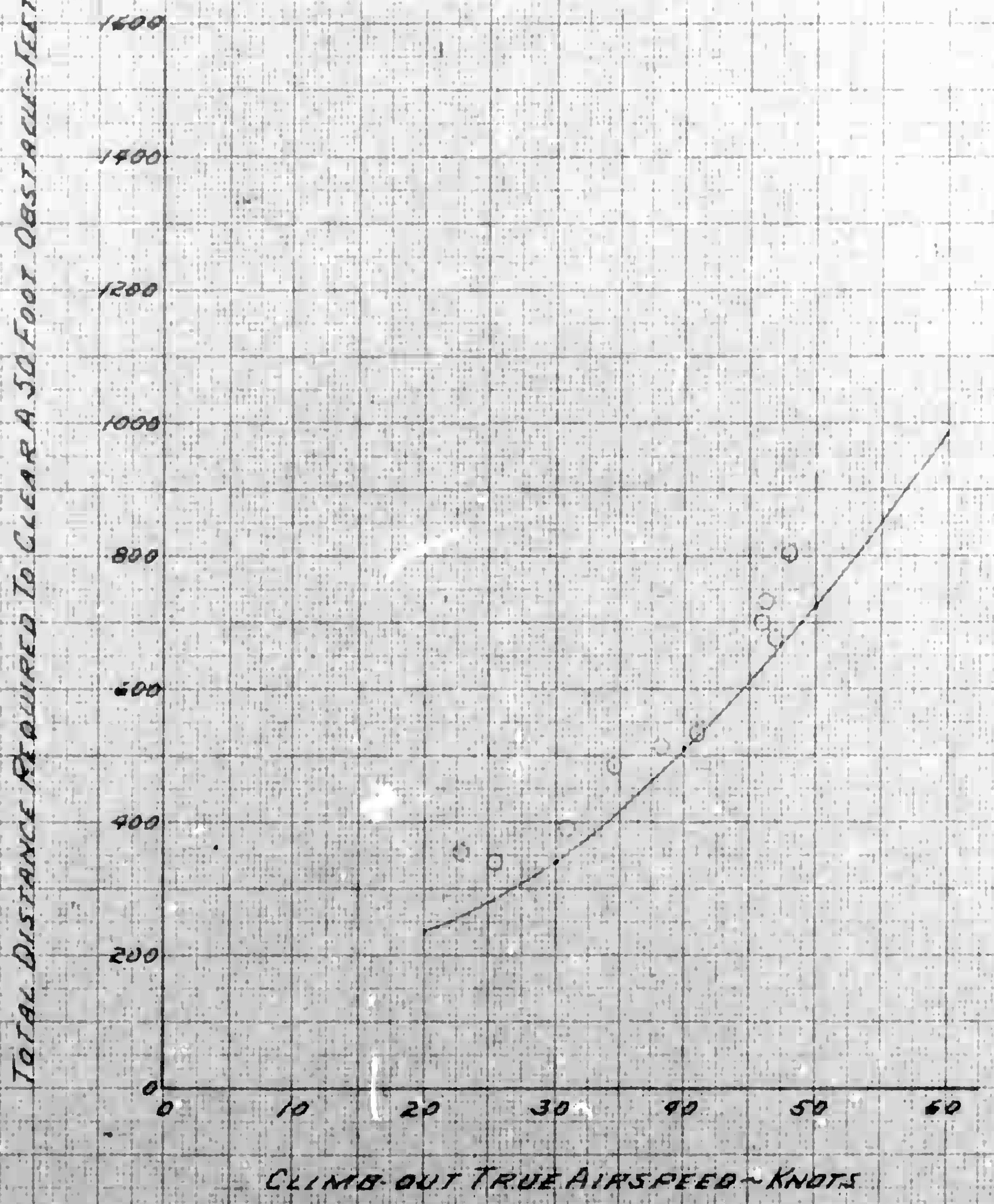

FIGURE NO. 7
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 394
GROSS WEIGHT = 2519 LB.
FREE AIR TEMP = 22 °C
PRESSURE ALTITUDE = 4110 FT.
WIND VELOCITY = ≤3 KNOTS
$\Delta C_p = 2.68 \times 10^{-5}$
TECHNIQUE — CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS

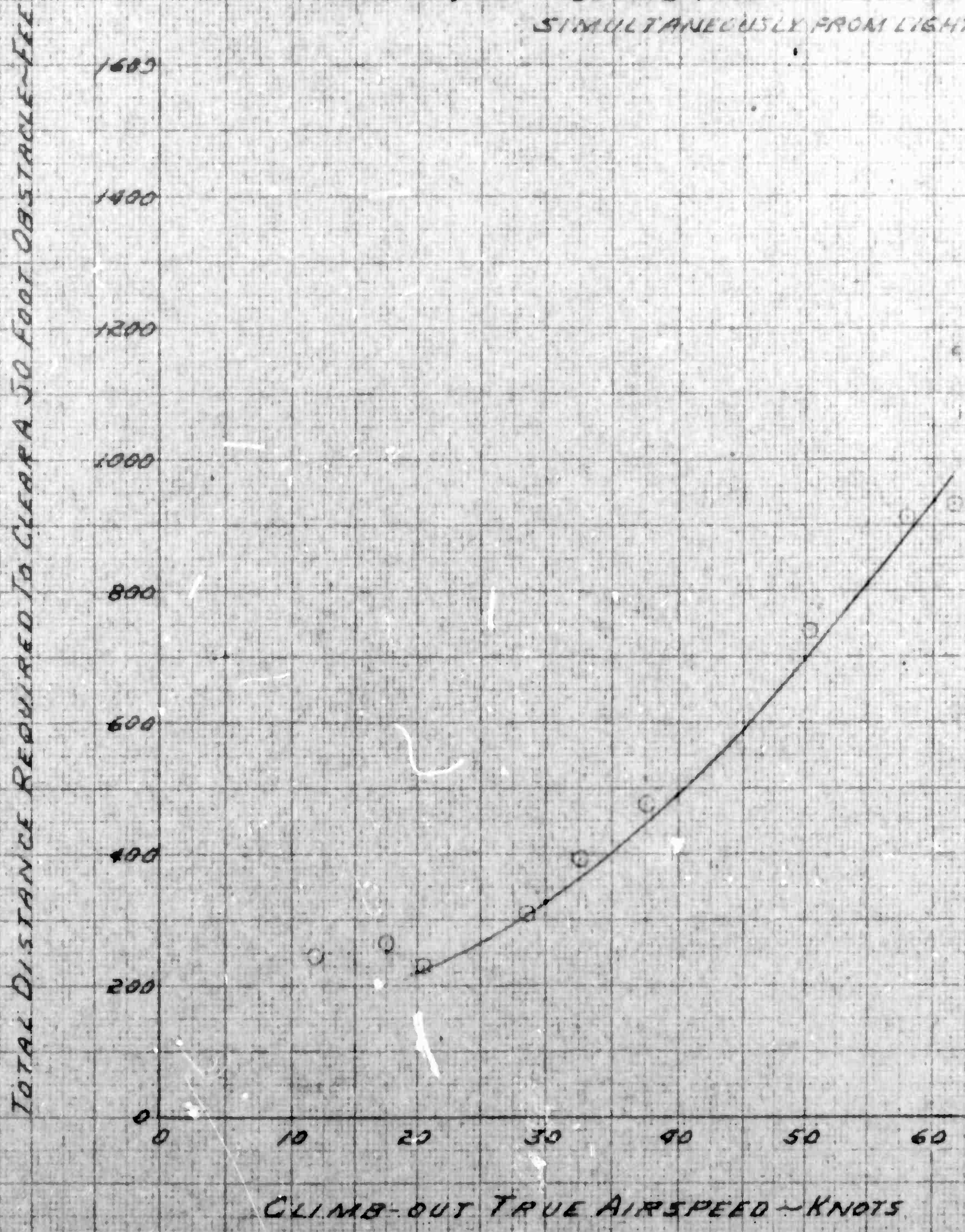

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FIGURE NO. 8
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 394
GROSS WEIGHT = 2547 LB.
FREE AIR TEMP = 10.5 °C
PRESSURE ALTITUDE = 3941 FT.
WIND VELOCITY = ≤3 KNOTS
$\Delta C_p = 2.77 \times 10^{-5}$
TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS

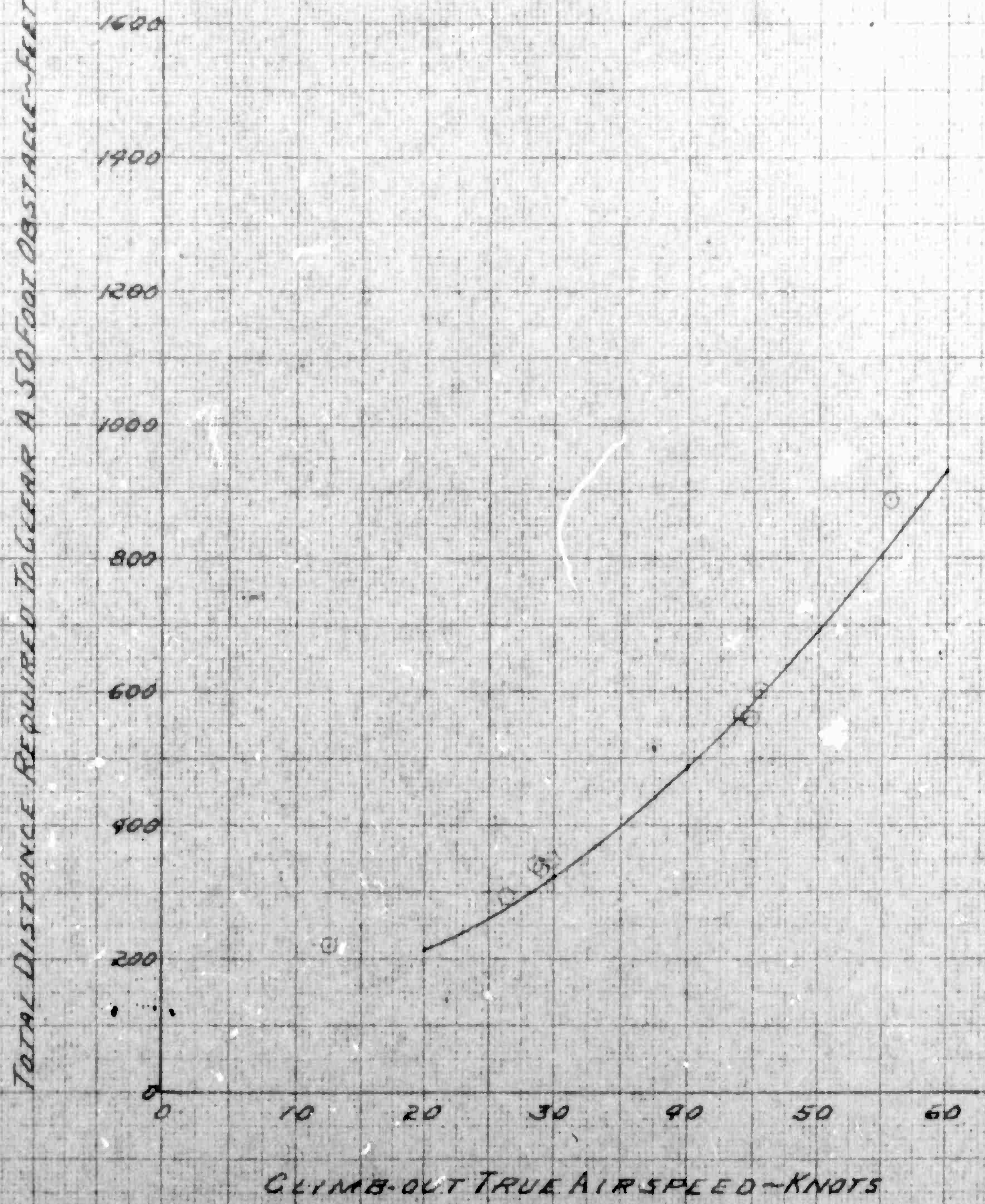


FIGURE NO. 9
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE

OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 394
GROSS WEIGHT = 2276 LB.
FREE AIR TEMP = 22.7 °C
PRESSURE ALTITUDE = 3940 FT.
WIND VELOCITY ≤ 3 KNOTS
AC_p = 3.44×10^{-5}
TECHNIQUE: CLIMB AND ACCELERATE SIMULTANEOUSLY FROM LIGHT ON SKIDS

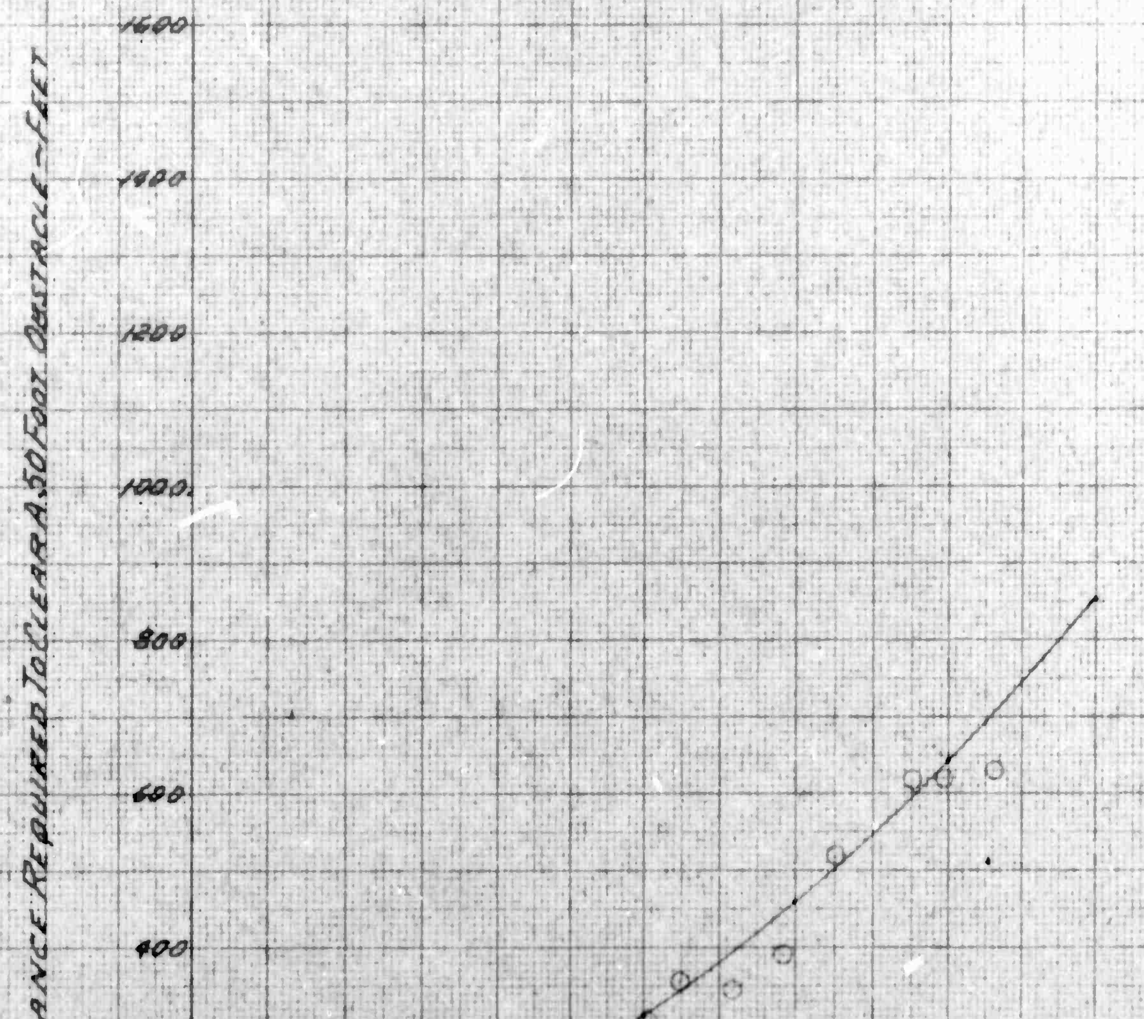


FIGURE NO. 10
SUMMARY TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION

TECHNIQUE: LEVEL ACCELERATION
FROM A 2 FOOT HOVER
GROSS WEIGHT = 2573 LB
ROTOR SPEED = 394 RPM
- - - - OUT OF GROUND EFFECT
HOVERING

NOTE:
DERIVED FROM FIGURES
12, 25 AND 55

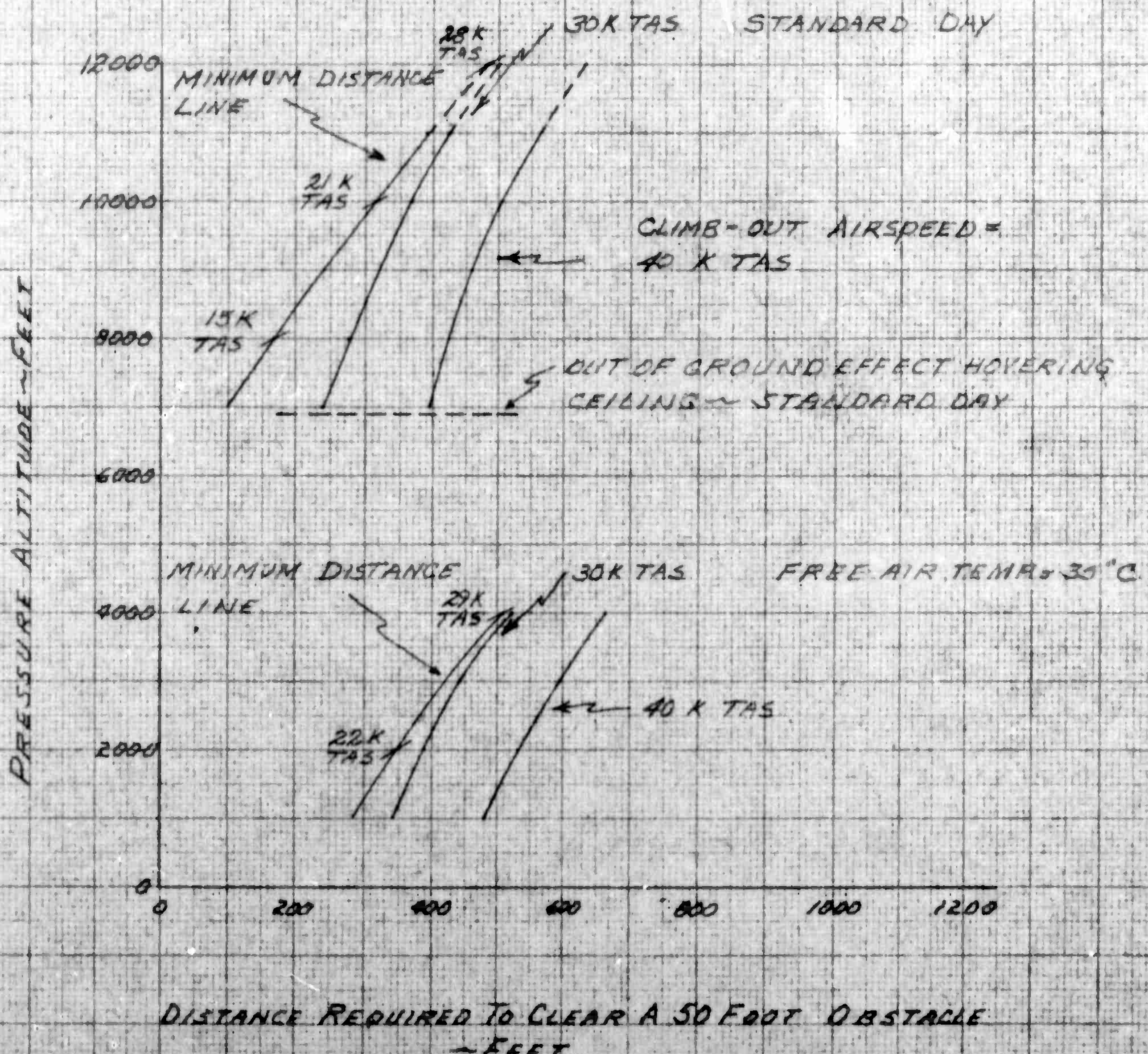


FIGURE NO. 11

SUMMARY TAKE-OFF DISTANCE REQUIRED

TO CLEAR A 50' OBSTACLE

OH-4A USA S/N 62-4202

CLEAN CONFIGURATION

TECHNIQUE LEVEL ACCELERATION
FROM A 2 FOOT HOVER
GROSS WEIGHT 2900 LB.
ROTOR SPEED 394 RPM
---- OUT OF GROUND EFFECT
HOVERING CEILING

NOTE:
DERIVED FROM FIGURES
12, 25 AND 55

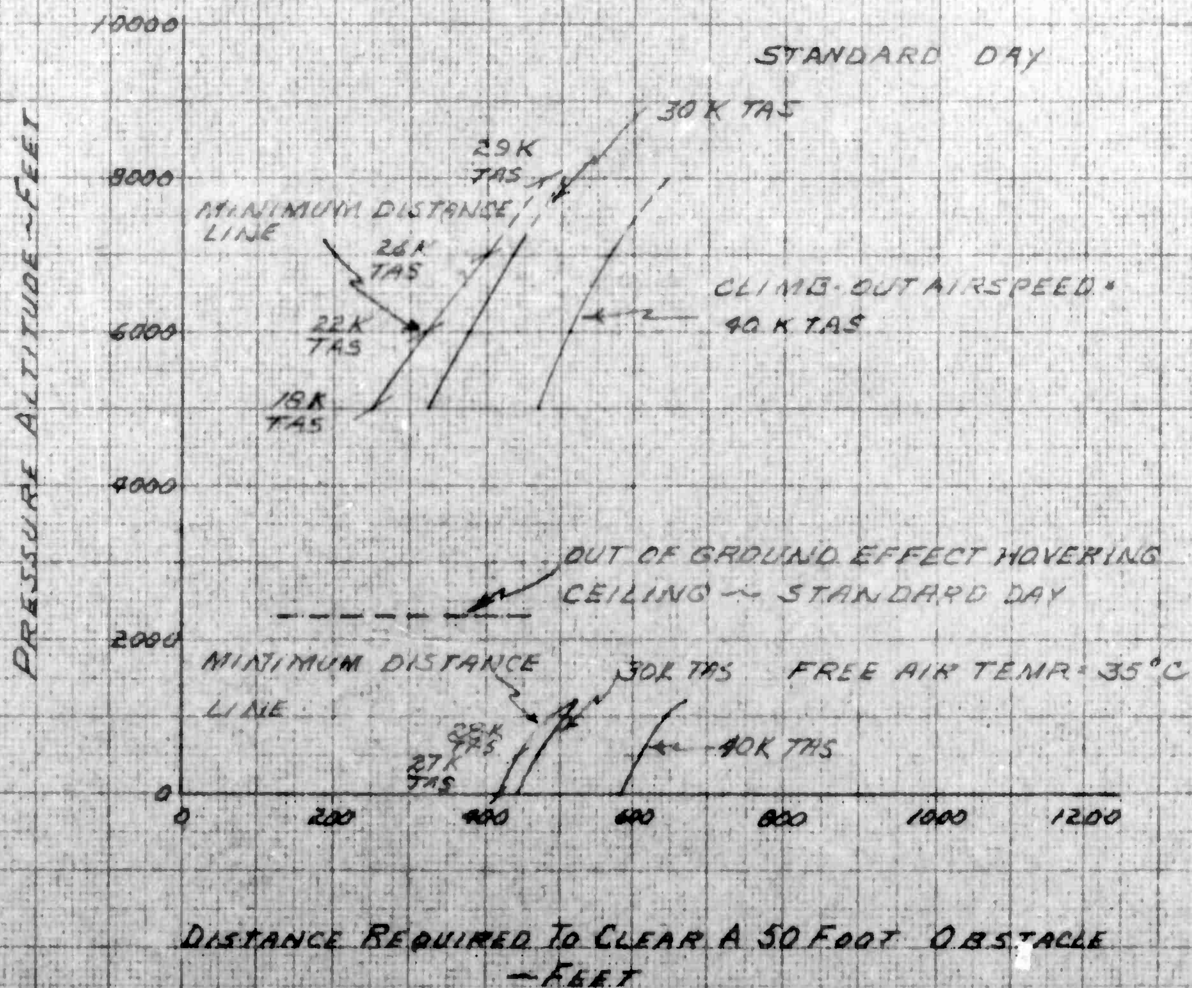


FIGURE No. 12

VARIATION OF TAKE-OFF DISTANCE
WITH CLIMB-OUT AIRSPEED AND ΔC_p

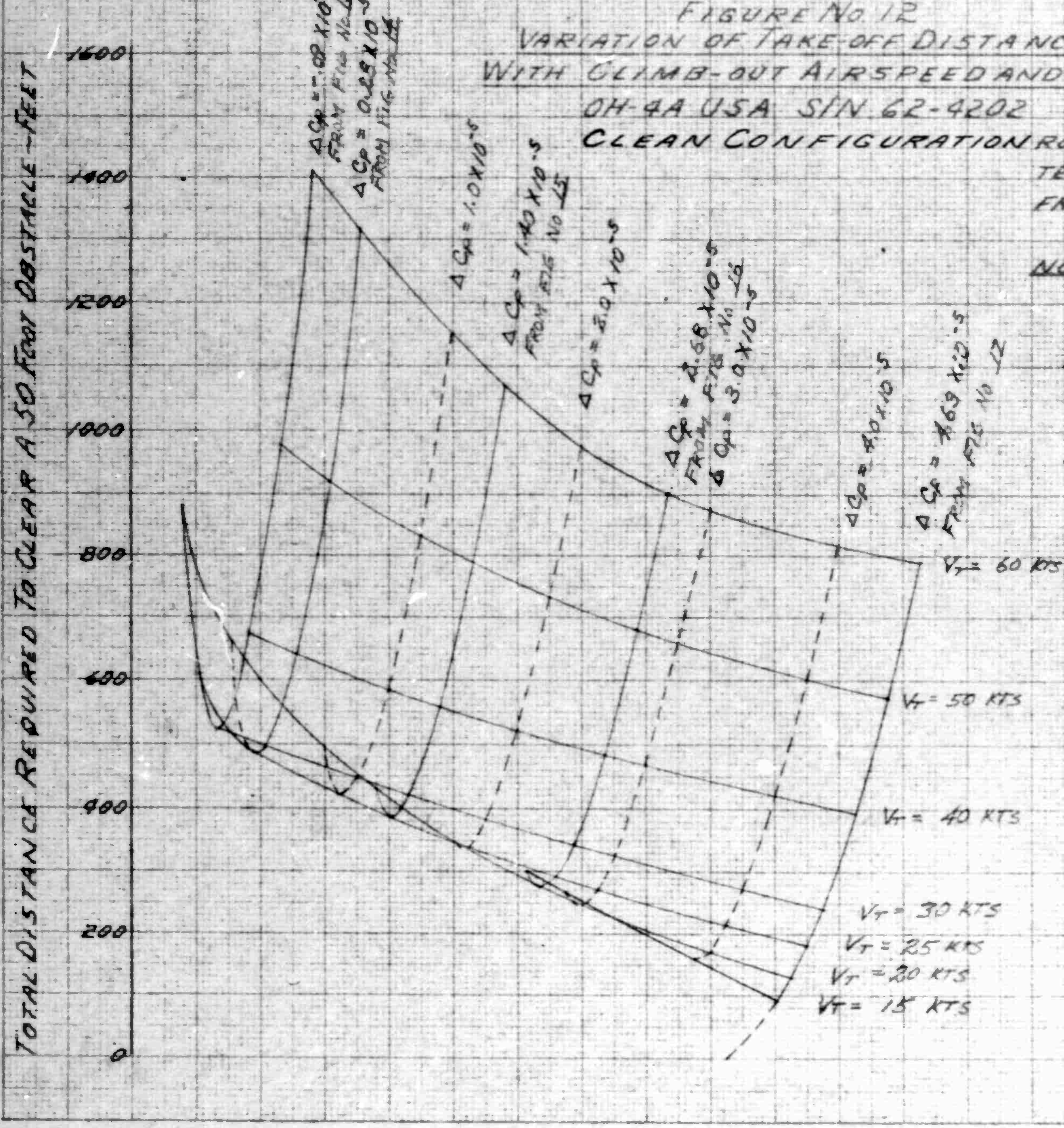
OH-4A USA S/N 62-4202

CLEAN CONFIGURATION ROTOR SPEED-354RPM

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER

NOTES.

1. ΔC_p: C_p AT TAKE-OFF - C_p TO HOVER AT 2 FEET.
2. C_p AT TAKE-OFF: BASED ON POWER AVAILABLE UNDER TAKE-OFF ATMOSPHERIC CONDITIONS.
3. C_p TO HOVER AT 2 FEET: BASED ON POWER REQUIRED TO HOVER AT 2 FEET UNDER TAKE-OFF ATMOSPHERIC CONDITIONS.



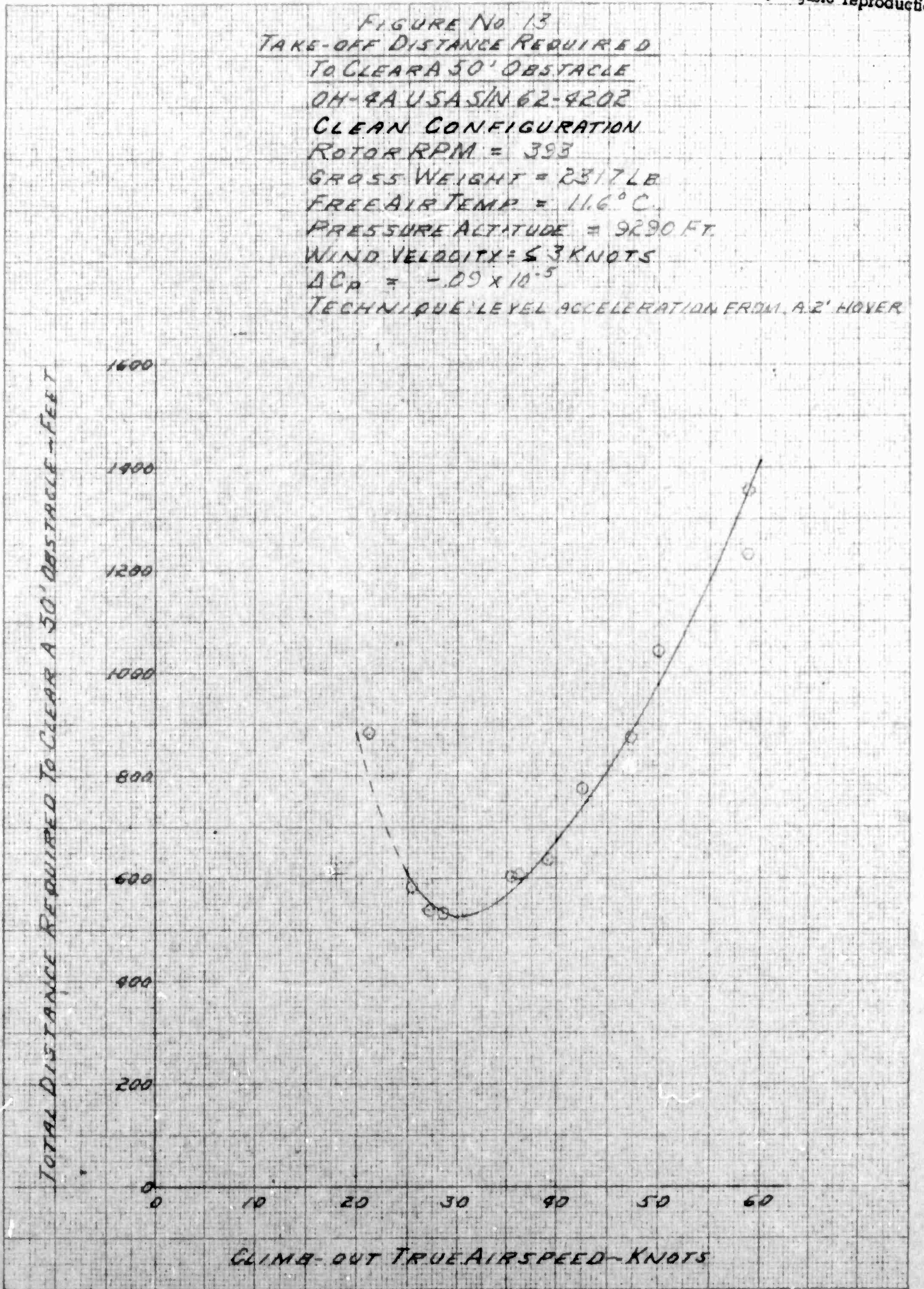
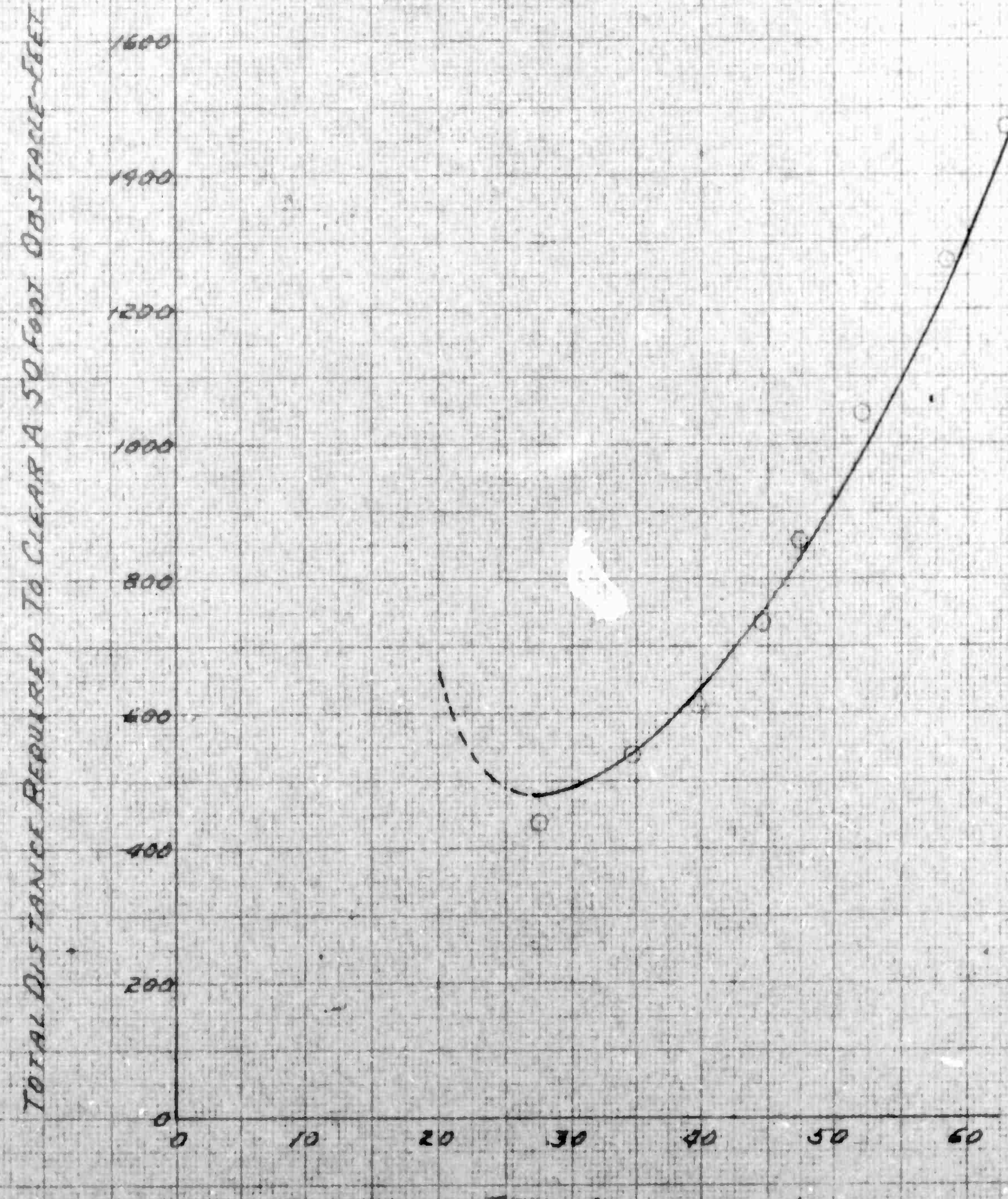


FIGURE NO 13
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 393
GROSS WEIGHT = 2317 LB
FREE AIR TEMP = 11.6°C
PRESSURE ALTITUDE = 9290 FT
WIND VELOCITY = ≤ 3 KNOTS
ΔCp = -.09 x 10⁻⁵
TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER
TOTAL DISTANCE REQUIRED TO CLEAR A 50' OBSTACLE ~ FEET
1600
1400
1200
1000
800
600
400
200
0
0
10
20
30
40
50
60
CLIMB-OUT TRUE AIRSPEED ~ KNOTS

FIGURE No 14
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 394
GROSS WEIGHT = 2440 LB
FREE AIR TEMP = 6.6 °C
PRESSURE ALTITUDE = 9450 FT
WIND VELOCITY = ≤ 3 KNOTS
ΔC_P = 0.28 x 10^{-5}
TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER



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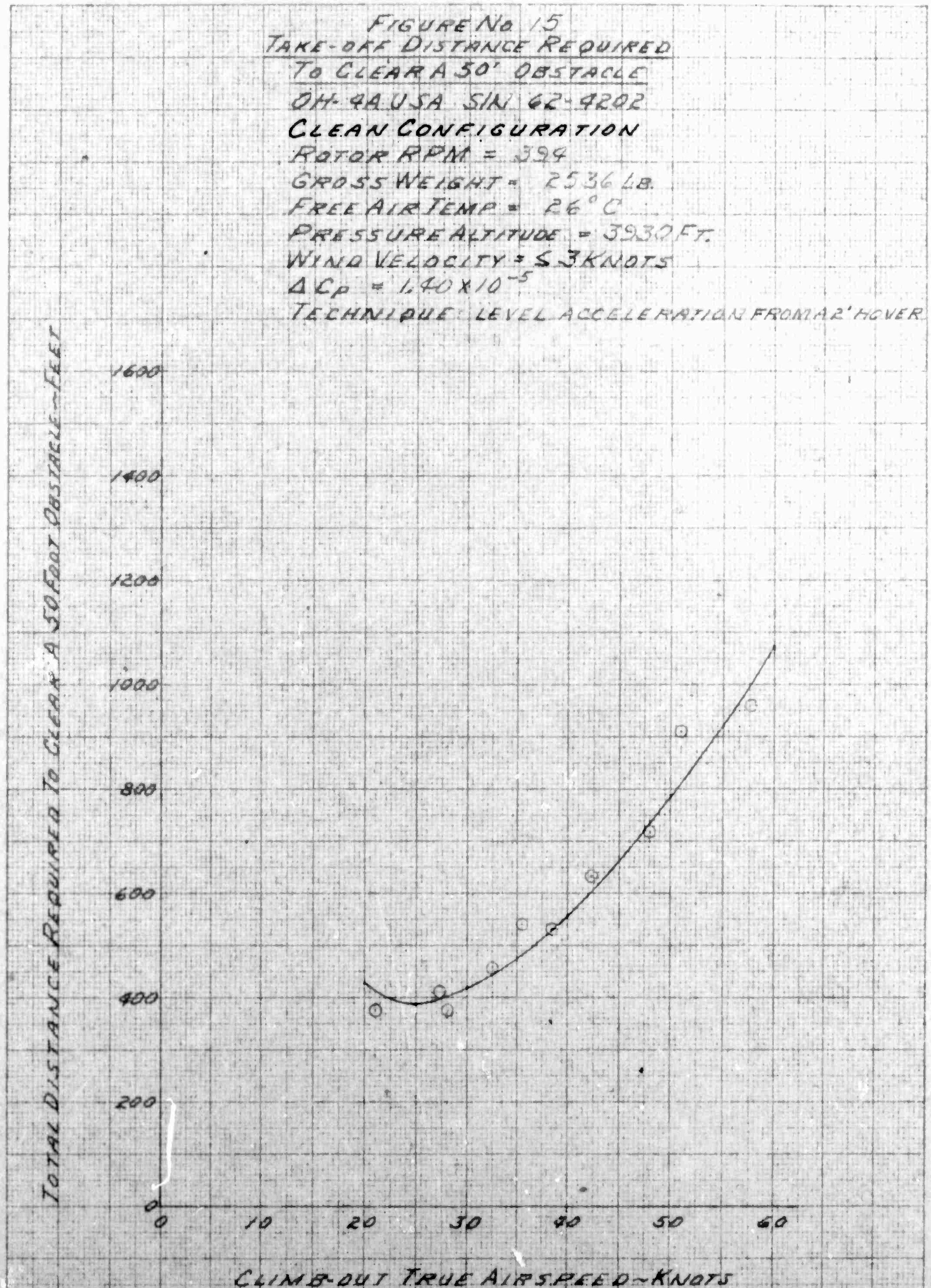

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FIGURE No. 16
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 394
GROSS WEIGHT = 2415 LB.
FREE AIR TEMP. = 22.9°C
PRESSURE ALTITUDE = 4120 FT.
WIND VELOCITY = ≤ 3 KNOTS
$\Delta C_P = 2.68 \times 10^{-5}$
TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER

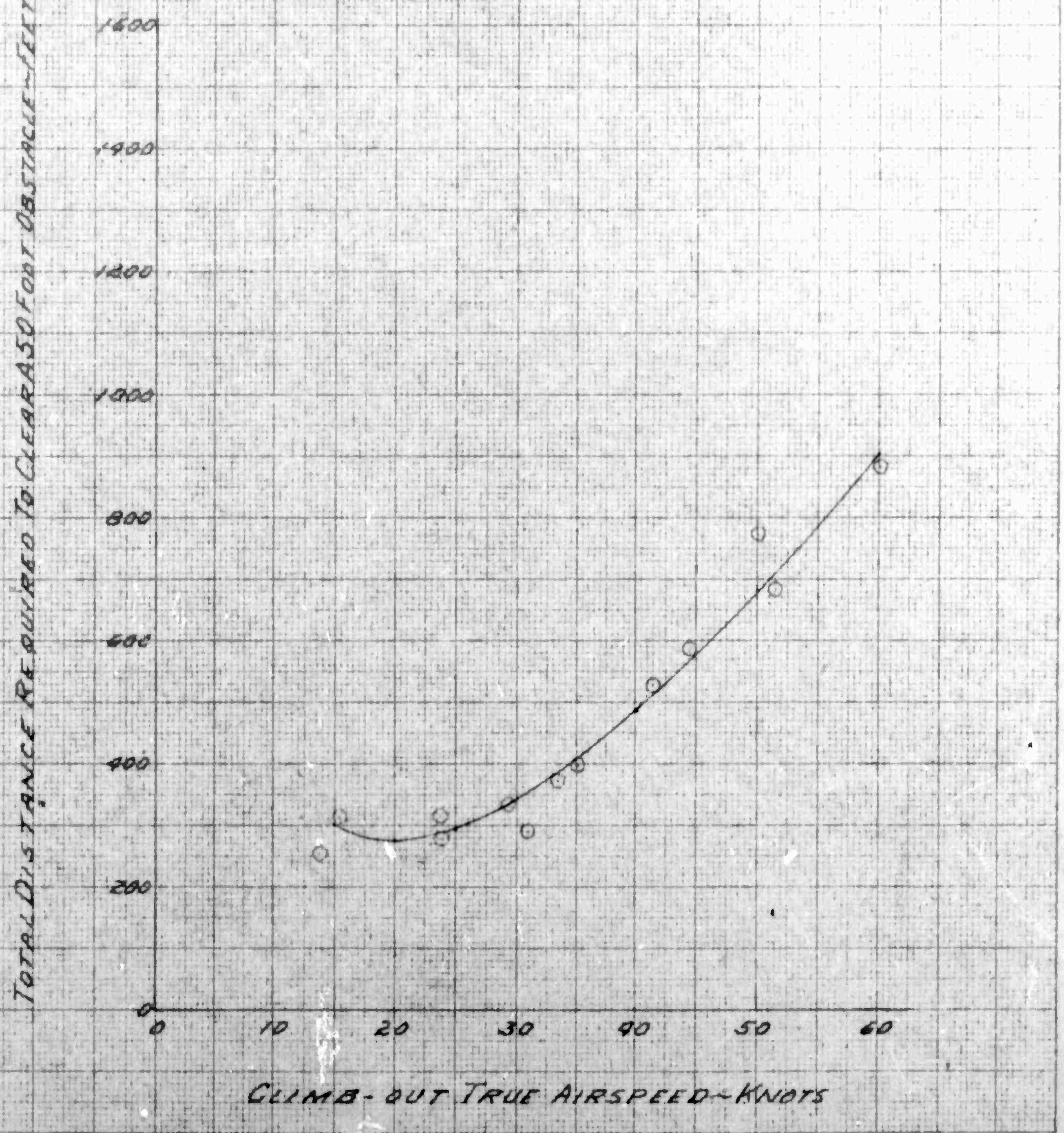

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FIGURE No 17
TAKE-OFF DISTANCE REQUIRED
TO CLEAR A 50' OBSTACLE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION
ROTOR RPM = 394
GROSS WEIGHT = 2254 LB.
FREE AIR TEMP = 14.4°C
PRESSURE ALTITUDE = 4300 FT.
WIND VELOCITY = ≤ 3 KNOTS
$\Delta C_P = 4.63 \times 10^{-5}$
TECHNIQUE: LEVEL ACCELERATION FROM A 2' HOVER

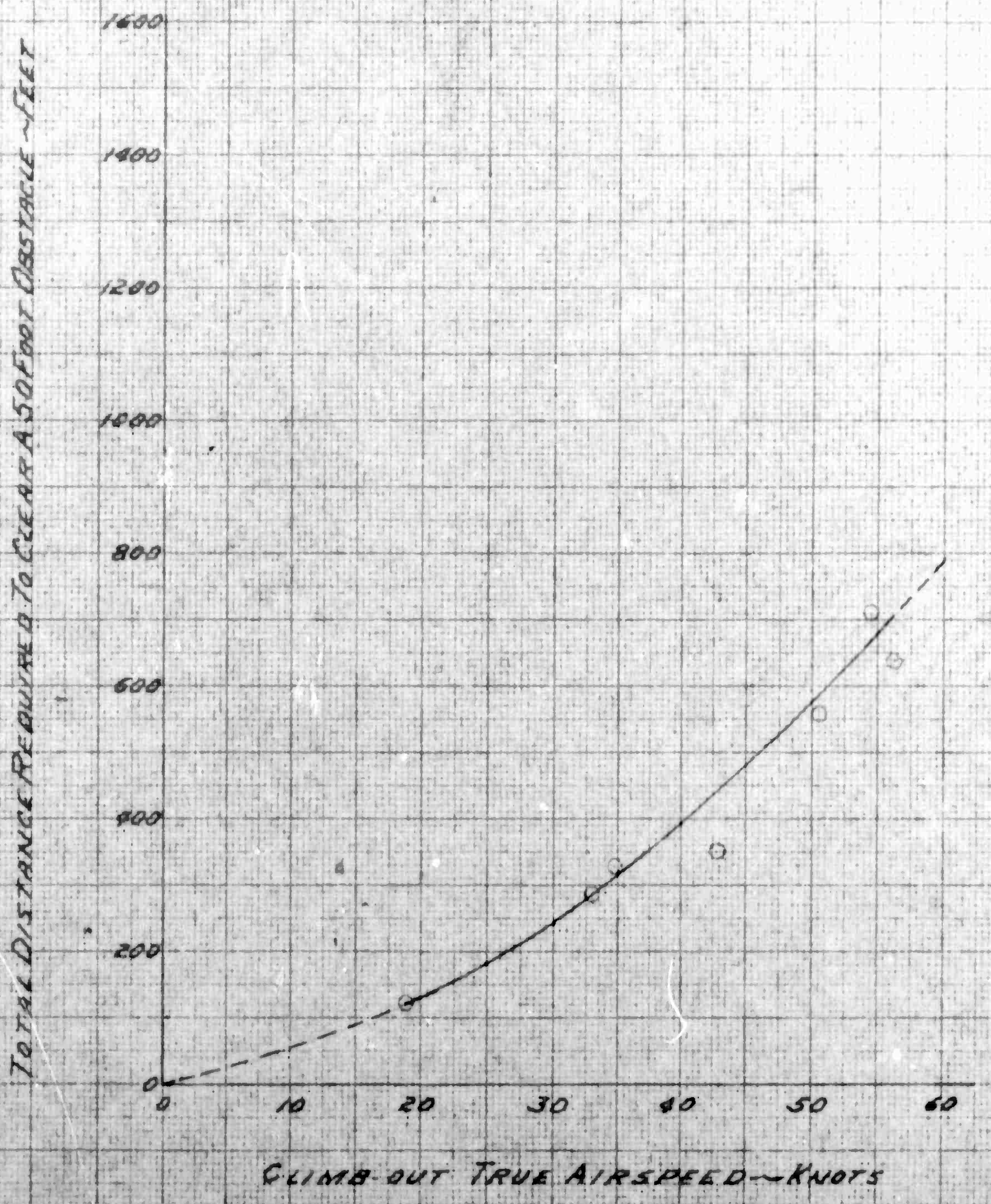

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FIGURE NO 18
SUMMARY HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
ROTOR SPEED: 394 RPM
TAKE-OFF POWER
CLEAN CONFIGURATION

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

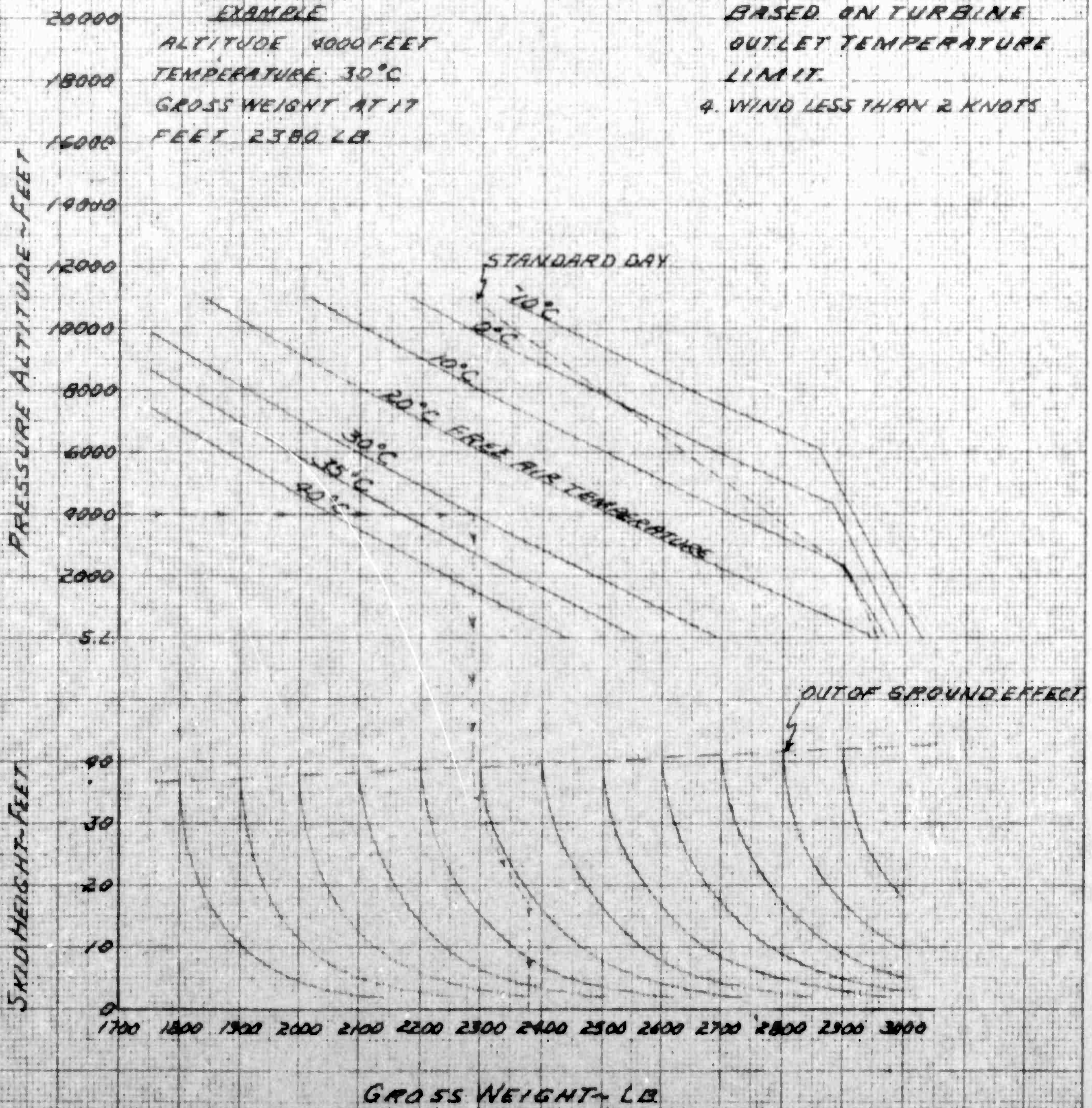

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FIGURE No. 19
SUMMARY HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
ROTOR SPEED - 394 RPM
MAXIMUM CONTINUOUS POWER
CLEAN CONFIGURATION

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

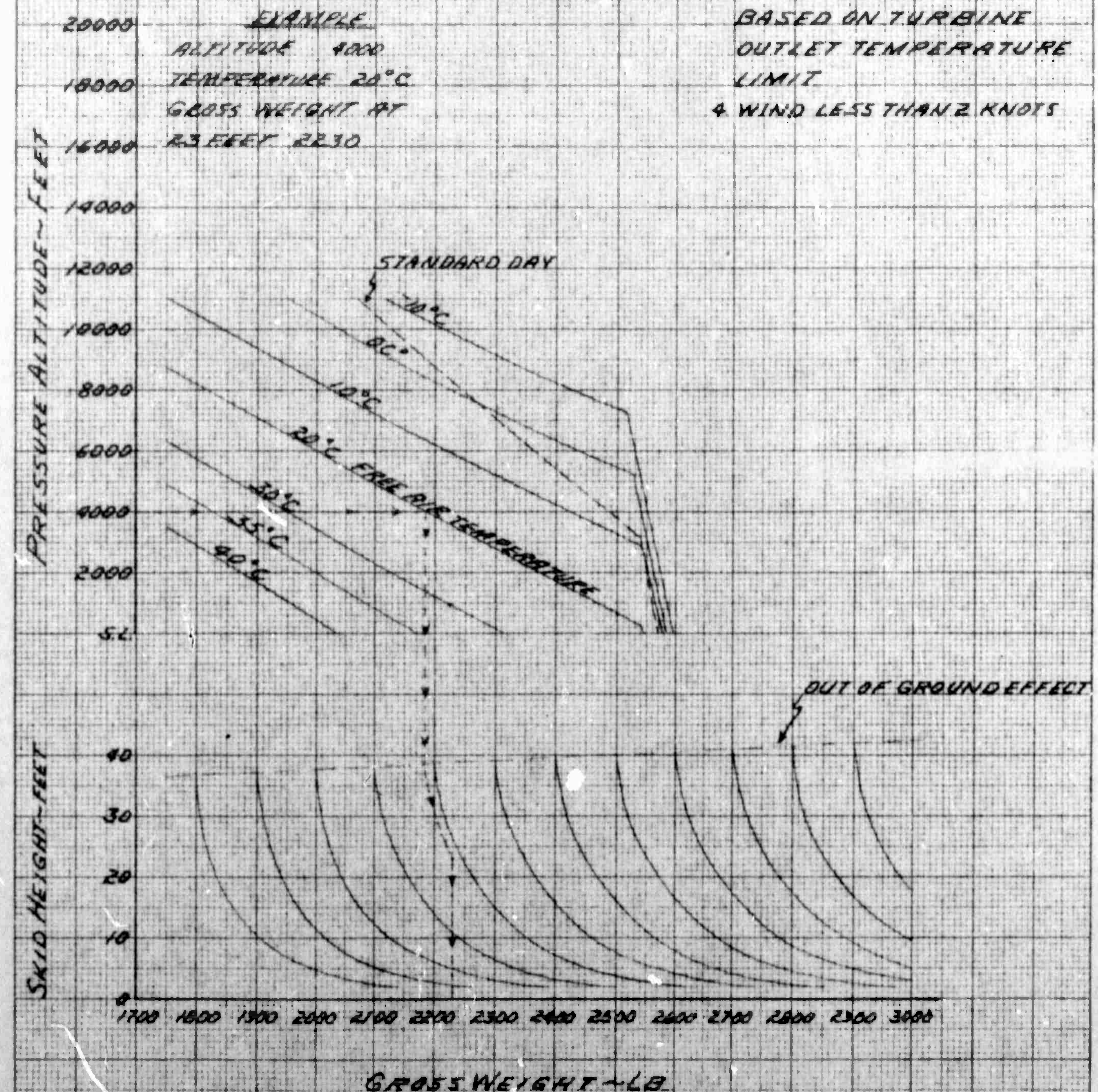

FIGURE NO. 20
SUMMARY HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
ROTOR SPEED - 394 RPM
TAKE-OFF POWER
CLEAN CONFIGURATION

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.

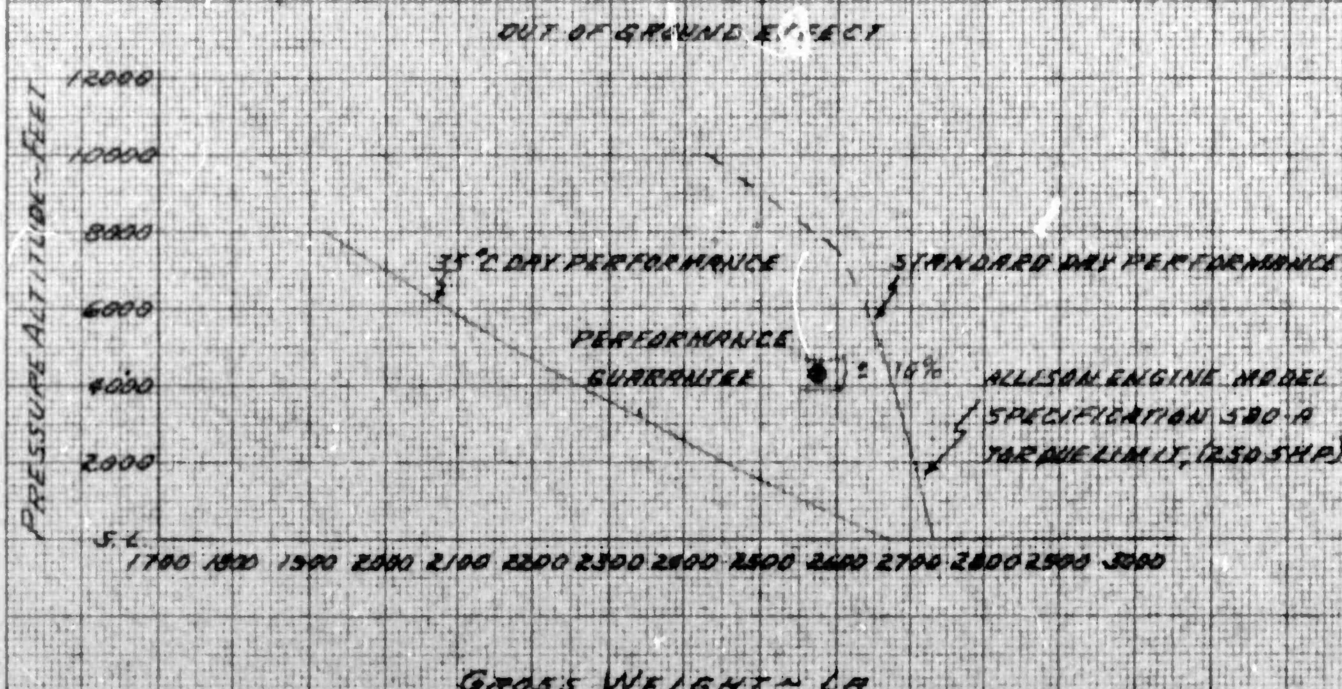

FIGURE NO 21
FUEL FLOW DURING HOVERING FLIGHT
OH-4A USA S/N 62-4202
U.S. STANDARD DAY

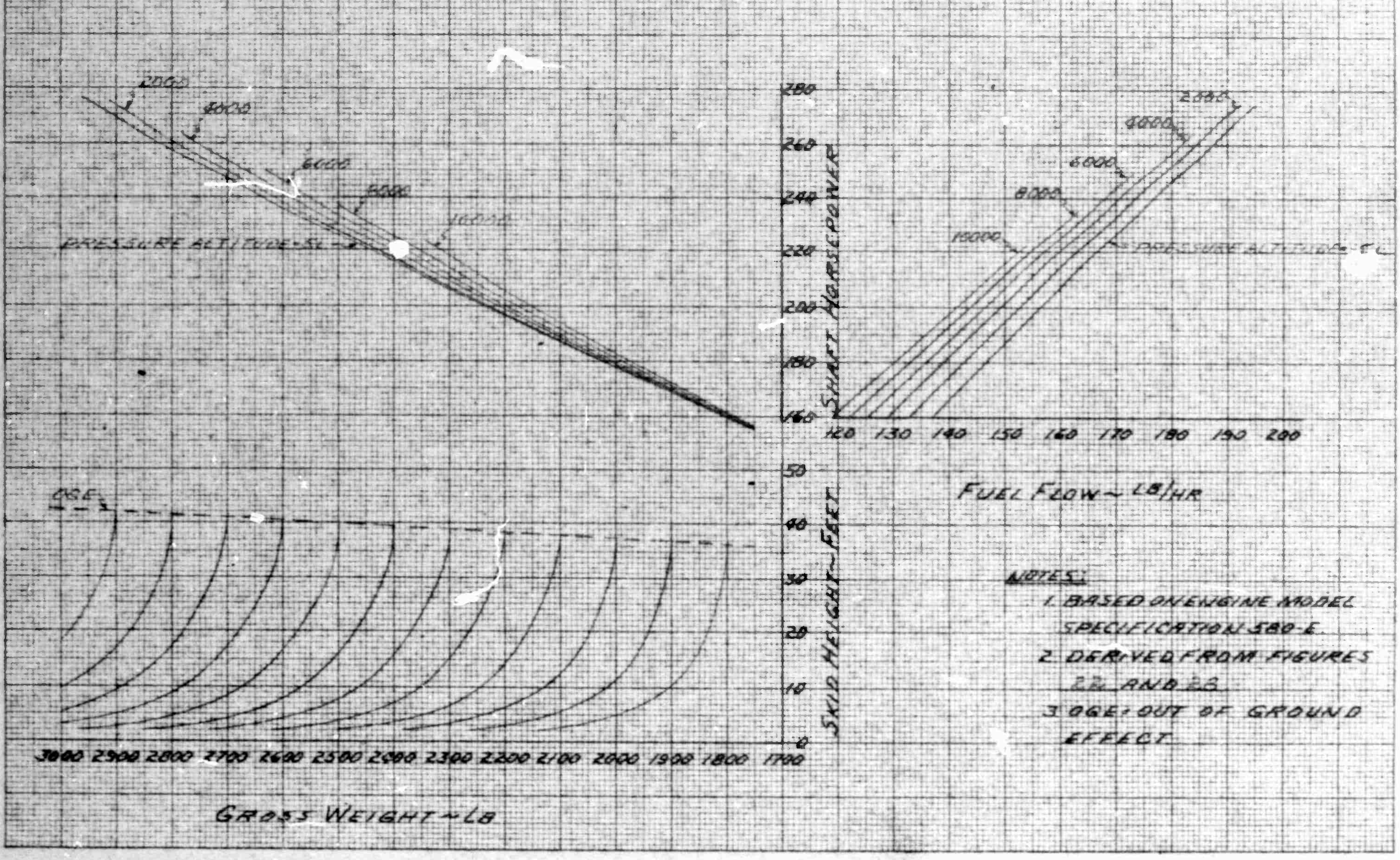

FIGURE NO 22
HOVERING PERFORMANCE
OH-4A U.S.A S/N 62-4202
FREE FLIGHT
CLEAN CONFIGURATION

NOTES:
1. DERIVED FROM FAIRED CURVES ON FIGURE 29
2. VERTICAL DISTANCE FROM BOTTOM OF SKIDS TO CENTER OF ROTOR HUB 8.5 FEET.

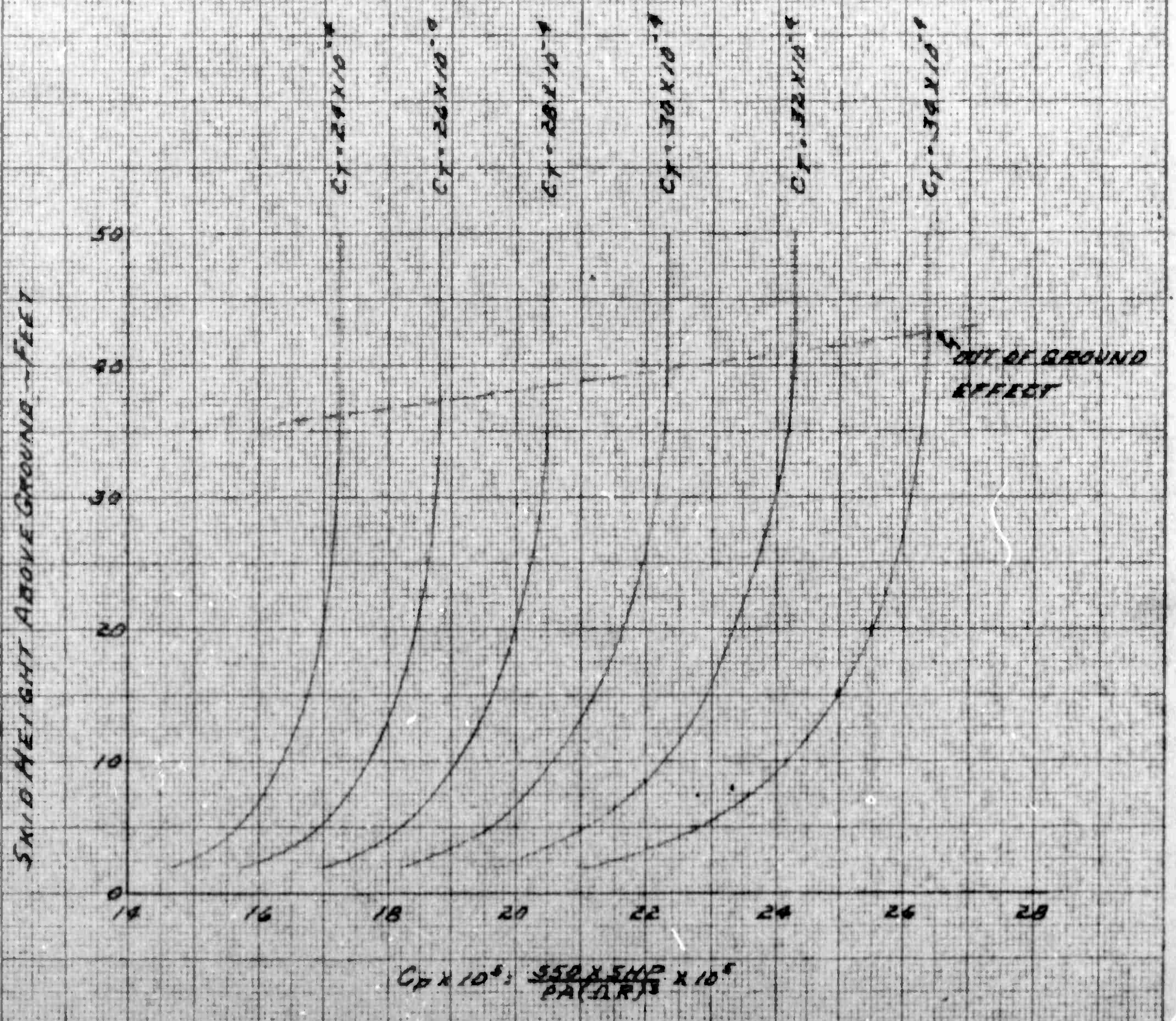

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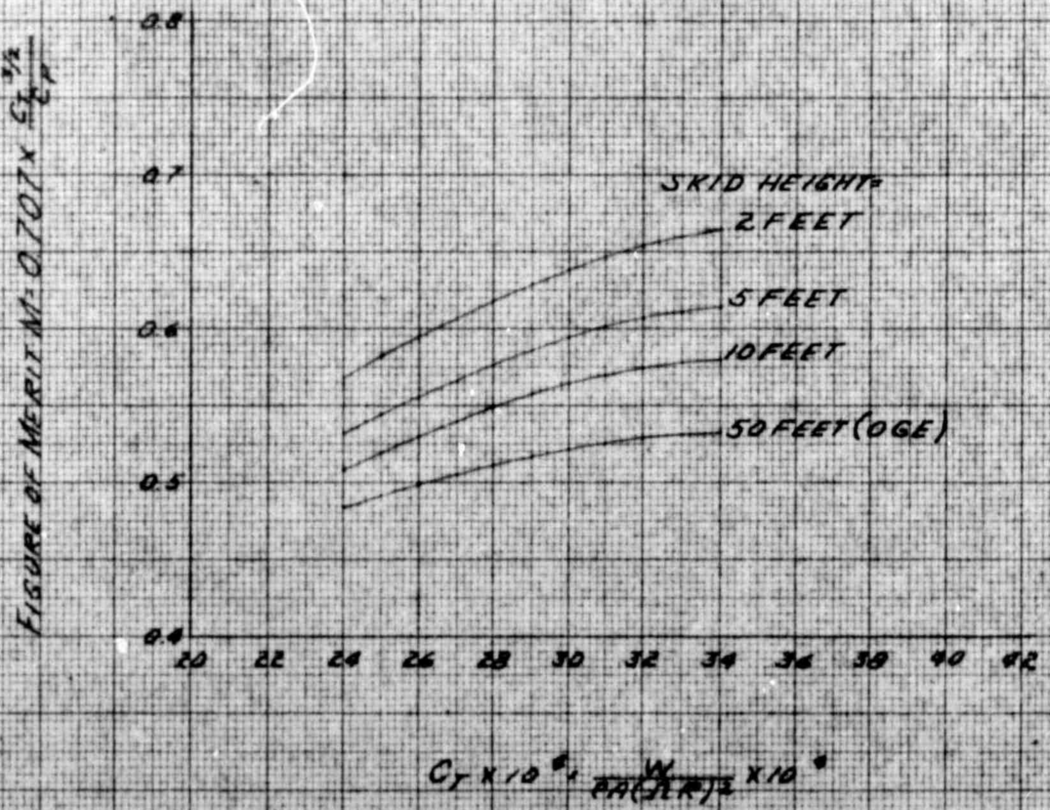
FIGURE NO 23
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
CLEAN CONFIGURATION

NOTE:
1. DERIVED FROM FIGURES 25 THROUGH 28
2. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.5 FEET
3. OGE = OUT OF GROUND EFFECT



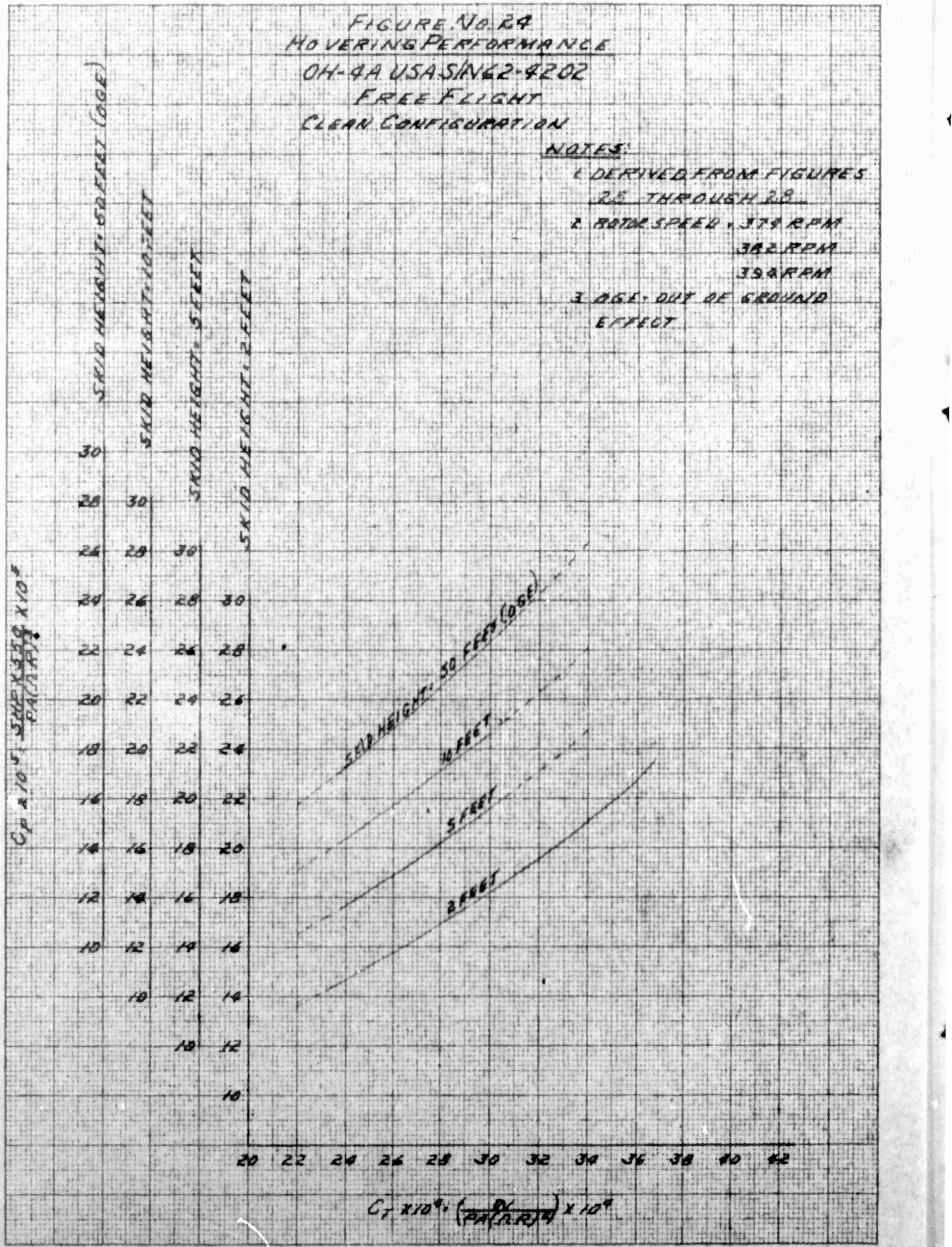


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FIGURE NO. 25
HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
FREE FLIGHT
SKID HEIGHT - 2 FEET
CLEAN CONFIGURATION

NOTES:
1. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.5 FEET.
2. WIND LESS THAN 2 KNOTS.

ROTOR SPEED-RPM 374	382	394	DENSITY ALTITUDE-FT.
○	○	○	190
□	□	□	5540
◇	◇	◇	6000
△	△	△	10000

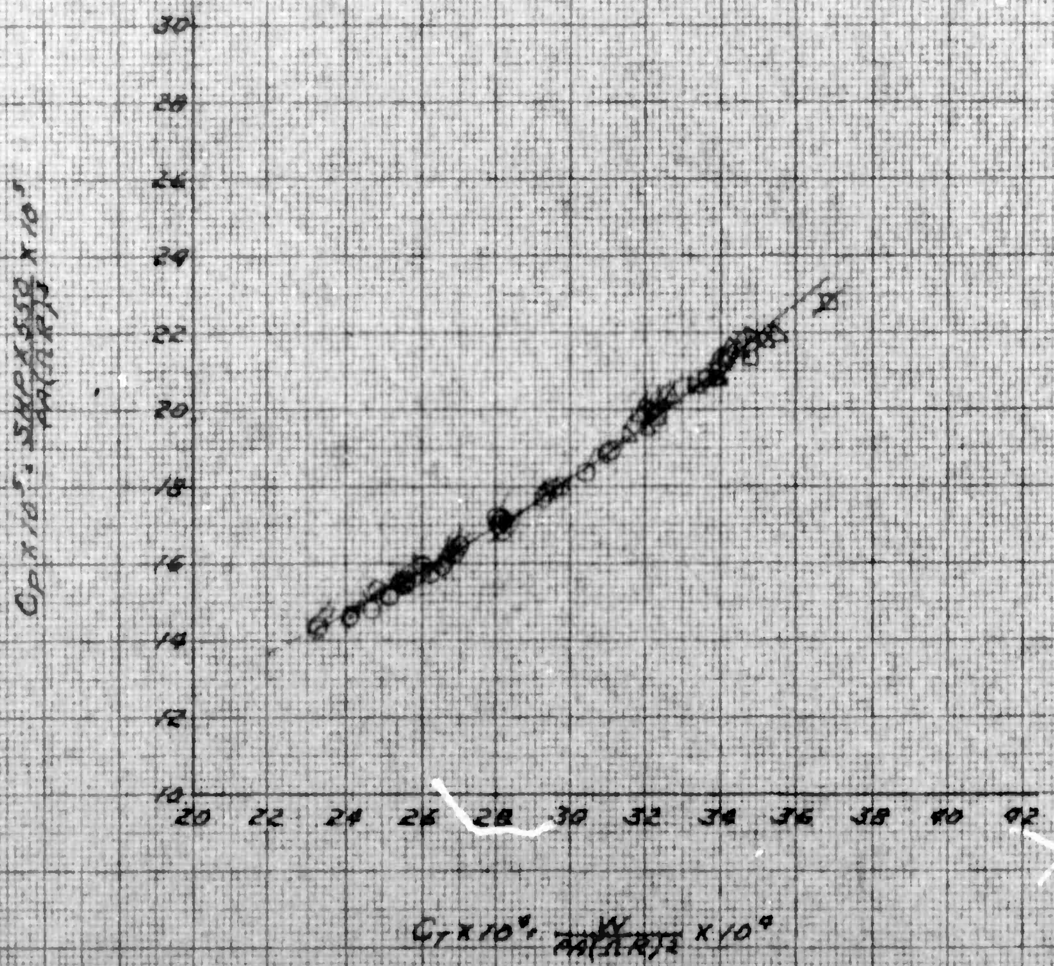


FIGURE No. 26
HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
FREE FLIGHT
SKID HEIGHT = 5 FEET
CLEAN CONFIGURATION

NOTES:
1. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 6.5 FEET
2. WIND LESS THAN 2 KNOTS

ROTOR SPEED-RPM			DENSITY ALTITUDE-FT.
379	382	394	
◻	⌀	○	140

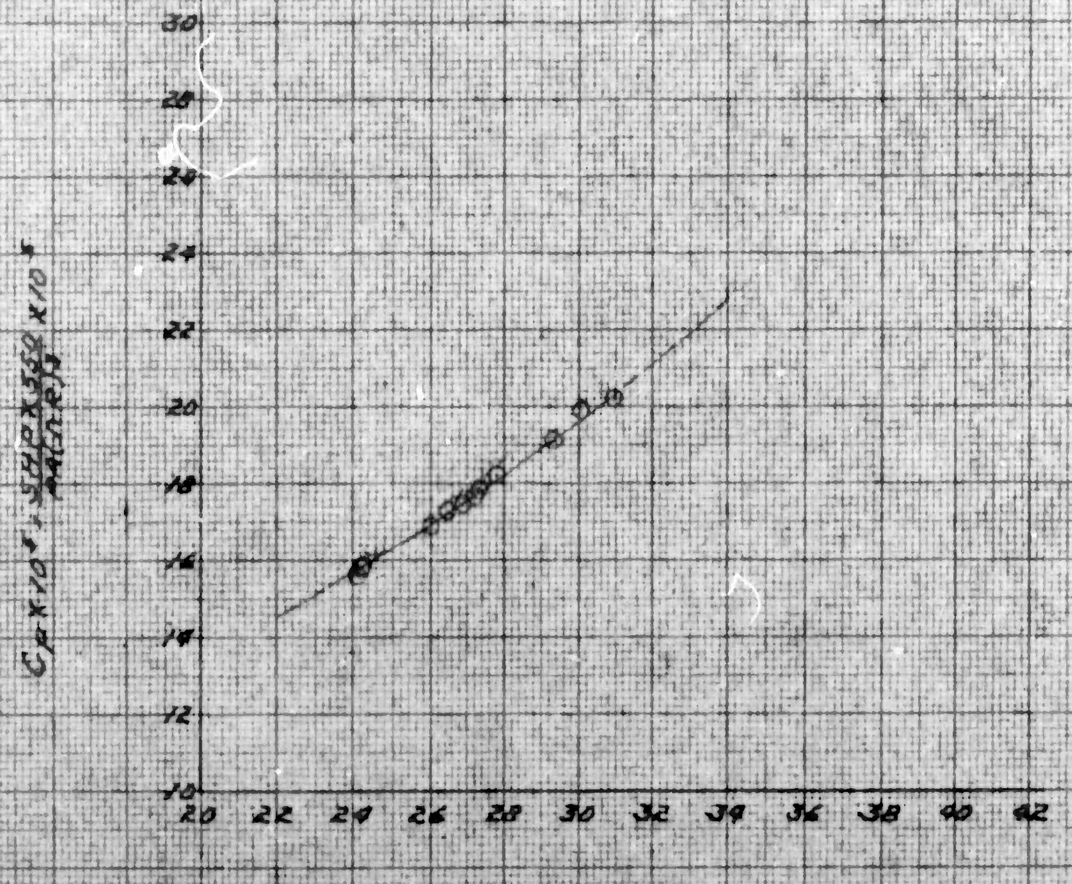
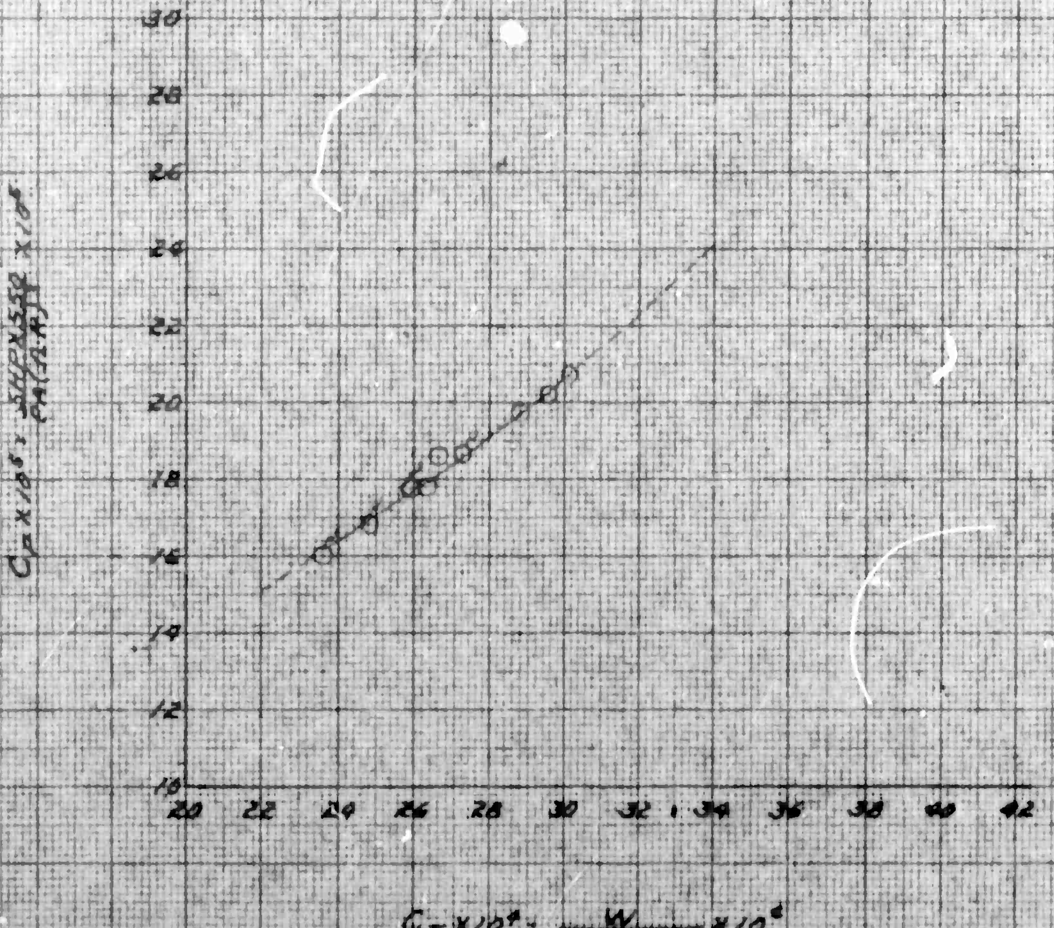


FIGURE NO 27
HOVERING PERFORMANCE
OH-4A USA S/N 62-4202
FREE FLIGHT
SKID HEIGHT = 1.0 FEET
CLEAN CONFIGURATION

NOTES:
1. VERTICAL DISTANCE FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB = 8.5 FEET
2. WIND LESS THAN 3 KNOTS

ROTOR SPEED-RPM			DENSITY ALTITUDE-FT
374	382	390	
○	○	○	140



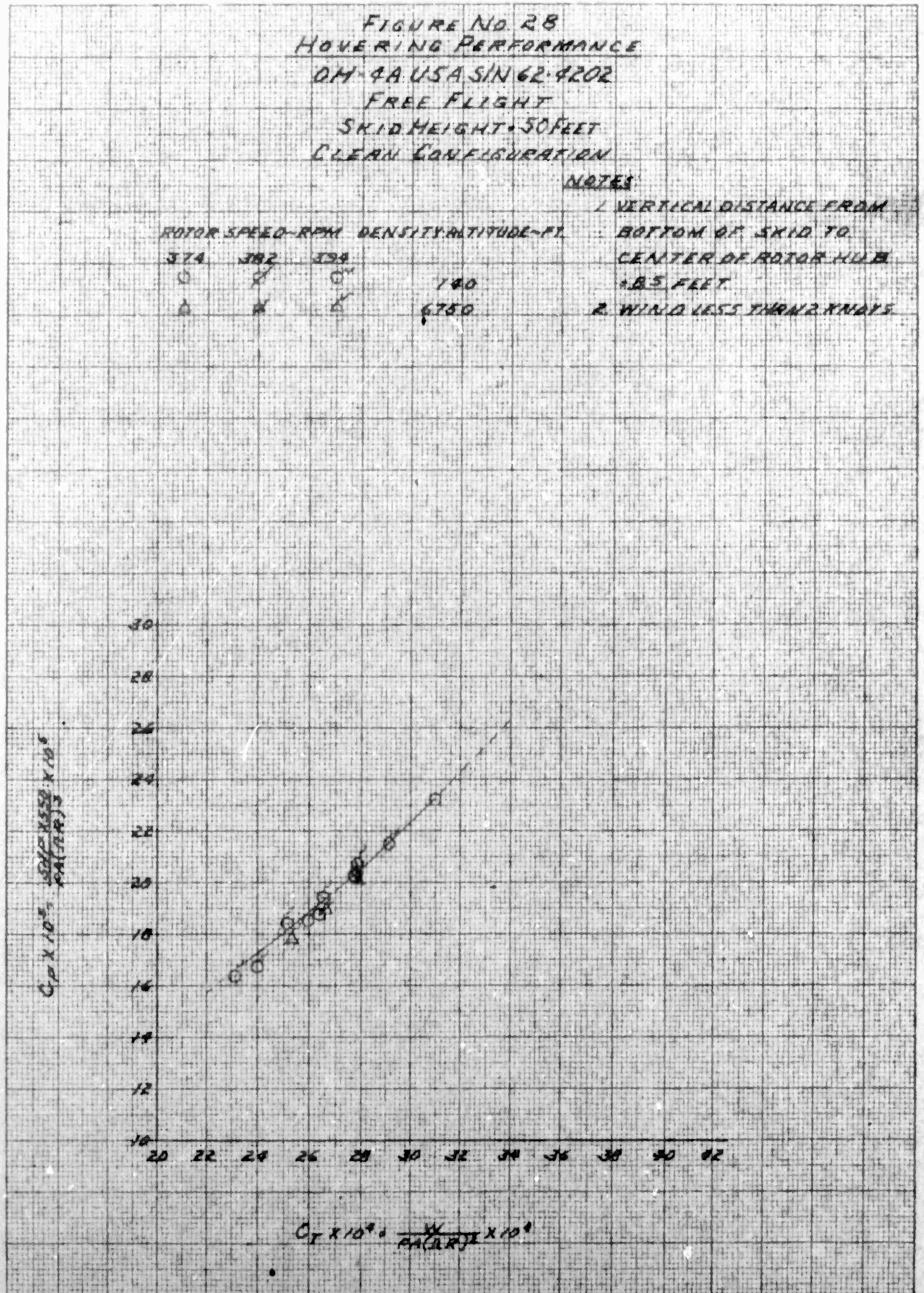


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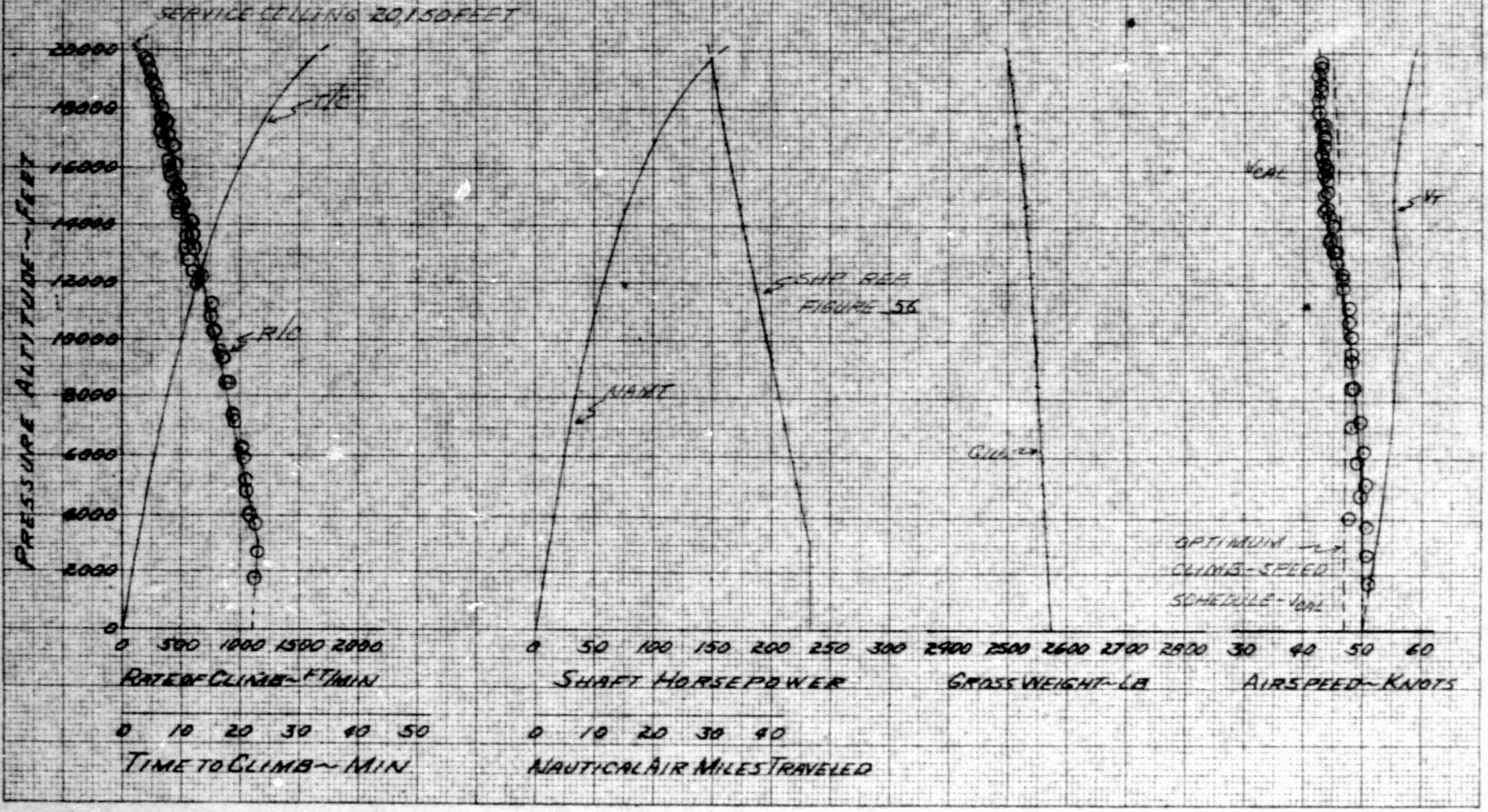
FIGURE NO. 29
CLIMB PERFORMANCE
OH-4A USA S/N 62-4202
MAXIMUM CONTINUOUS POWER
ROTOR SPEED=394 RPM
C.G.=STATION 102.0 (MID)
U.S. STANDARD DAY
CLEAN CONFIGURATION



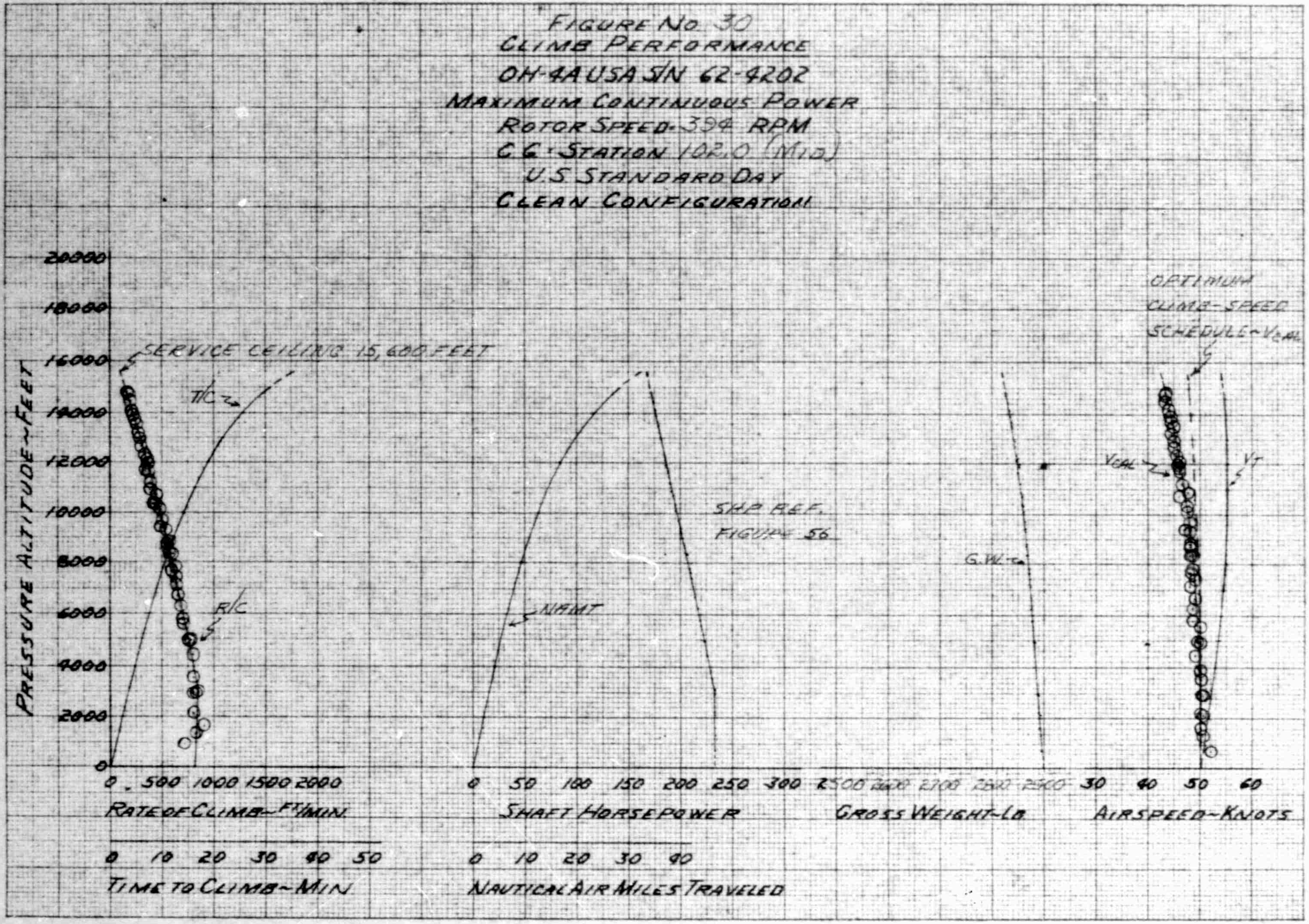


FIGURE NO. 30
CLIMB PERFORMANCE
OH-4A USA S/N 62-4202
MAXIMUM CONTINUOUS POWER
ROTOR SPEED-394 RPM
C.G. STATION 102.0 (MID)
U.S. STANDARD DAY
CLEAN CONFIGURATION
SERVICE CEILING 15,600 FEET
T/C
R/C
NAMT
SHP REF. FIGURE 56
G.W.
OPTIMUM CLIMB-SPEED SCHEDULE ~ V_{CAL}
V_{CAL}
V_T
PRESSURE ALTITUDE ~ FEET
0
2000
4000
6000
8000
10000
12000
14000
16000
18000
20000
0 500 1000 1500 2000
RATE OF CLIMB ~ FT/MIN
0 10 20 30 40 50
TIME TO CLIMB ~ MIN
0 50 100 150 200 250 300
SHAFT HORSEPOWER
0 10 20 30 40
NAUTICAL AIR MILES TRAVELED
2500 2600 2700 2800 2900
GROSS WEIGHT ~ LB
30 40 50 60
AIRSPEED ~ KNOTS

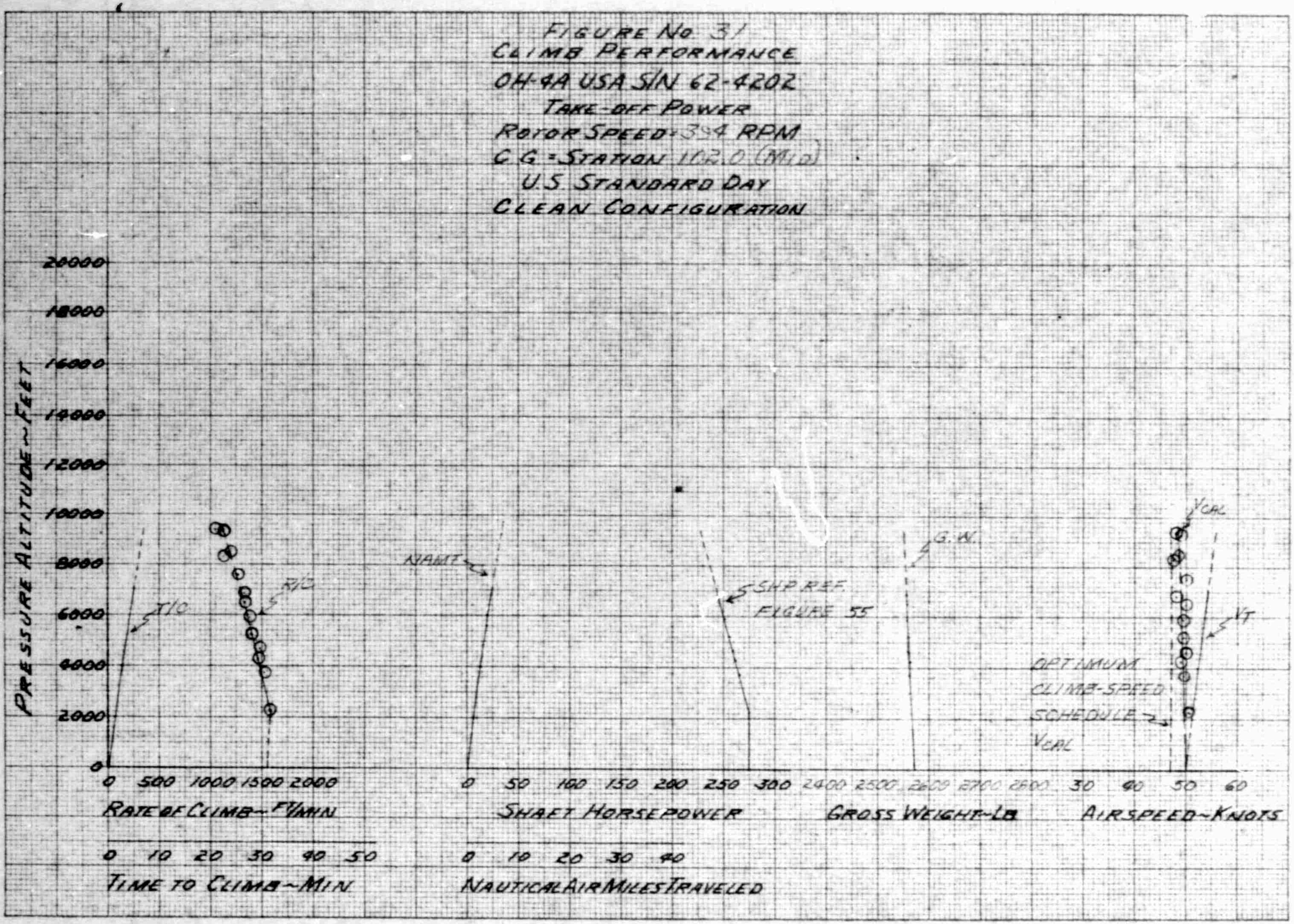


FIGURE No. 31
CLIMB PERFORMANCE
OH-4A USA S/N 62-4202
TAKE-OFF POWER
ROTOR SPEED = 354 RPM
C.G. = STATION 102.0 (MID)
U.S. STANDARD DAY
CLEAN CONFIGURATION
PRESSURE ALTITUDE ~ FEET
0
2000
4000
6000
8000
10000
12000
14000
16000
18000
20000
T/C
R/C
NAMT
SHP REF. FIGURE 55
G.W.
VCAL
VT
OPTIMUM CLIMB-SPEED SCHEDULE
VCAL
0 500 1000 1500 2000
RATE OF CLIMB ~ FT/MIN
0 10 20 30 40 50
TIME TO CLIMB ~ MIN
0 50 100 150 200 250 300
SHAFT HORSEPOWER
0 10 20 30 40
NAUTICAL AIR MILES TRAVELED
2400 2500 2600 2700 2800
GROSS WEIGHT ~ LB
30 40 50 60
AIRSPEED ~ KNOTS

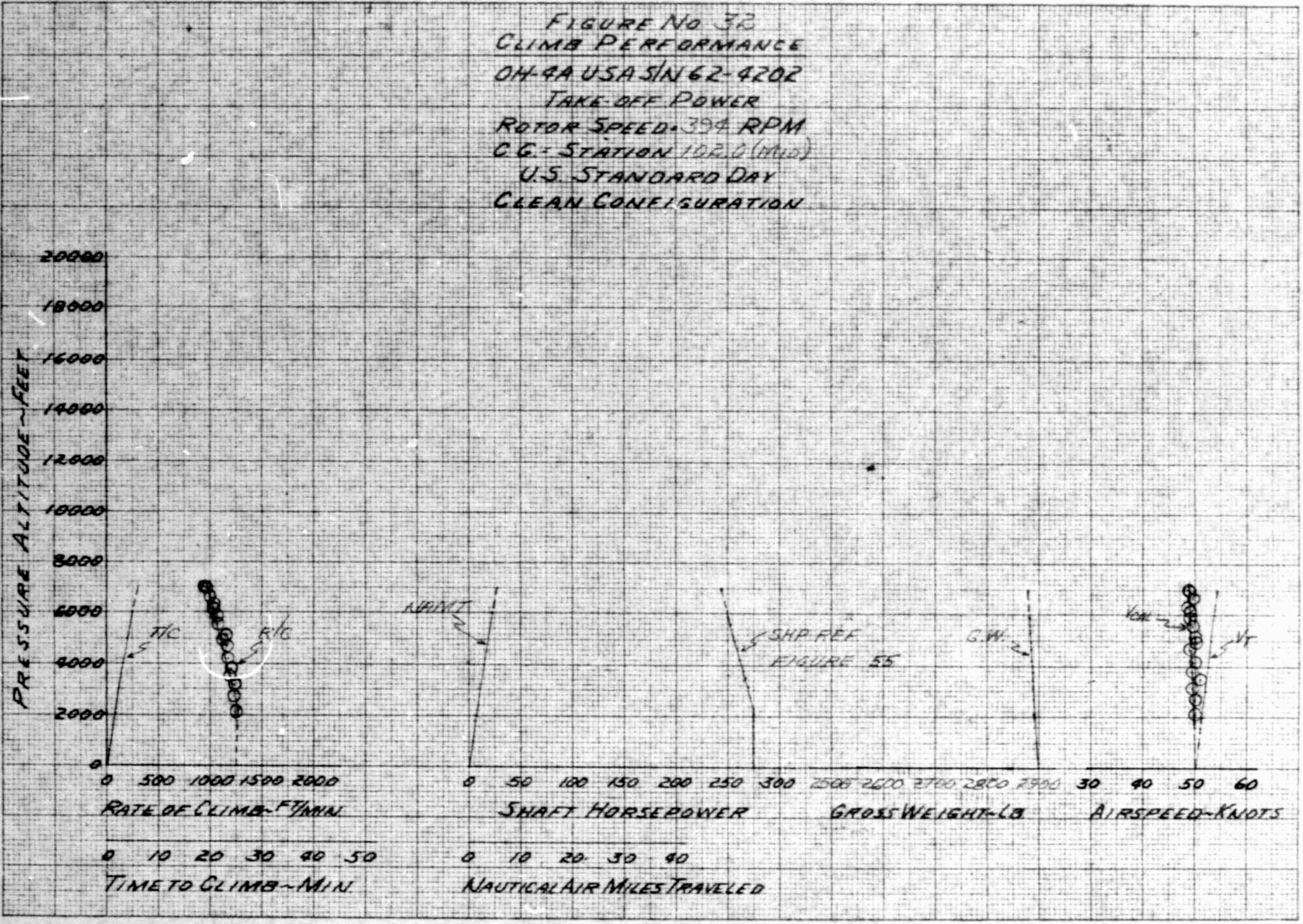


FIGURE NO 32
CLIMB PERFORMANCE
OH-4A USA S/N 62-4202
TAKE-OFF POWER
ROTOR SPEED- 394 RPM
C.G.- STATION 102.0 (MID)
U.S. STANDARD DAY
CLEAN CONFIGURATION
PRESSURE ALTITUDE ~ FEET
0
2000
4000
6000
8000
10000
12000
14000
16000
18000
20000
T/C
R/C
NAMT
SHP REF
FIGURE 55
G.W.
VCAL
VT
0 500 1000 1500 2000
RATE OF CLIMB ~ FT/MIN
0 10 20 30 40 50
TIME TO CLIMB ~ MIN
0 50 100 150 200 250 300
SHAFT HORSEPOWER
0 10 20 30 40
NAUTICAL AIR MILES TRAVELED
2500 2600 2700 2800 2900
GROSS WEIGHT ~ LB
30 40 50 60
AIRSPEED ~ KNOTS

FIGURE No 33

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

OH-4A USA S/N 62-4202

CLEAN CONFIGURATION

ROTOR SPEED: 394 RPM

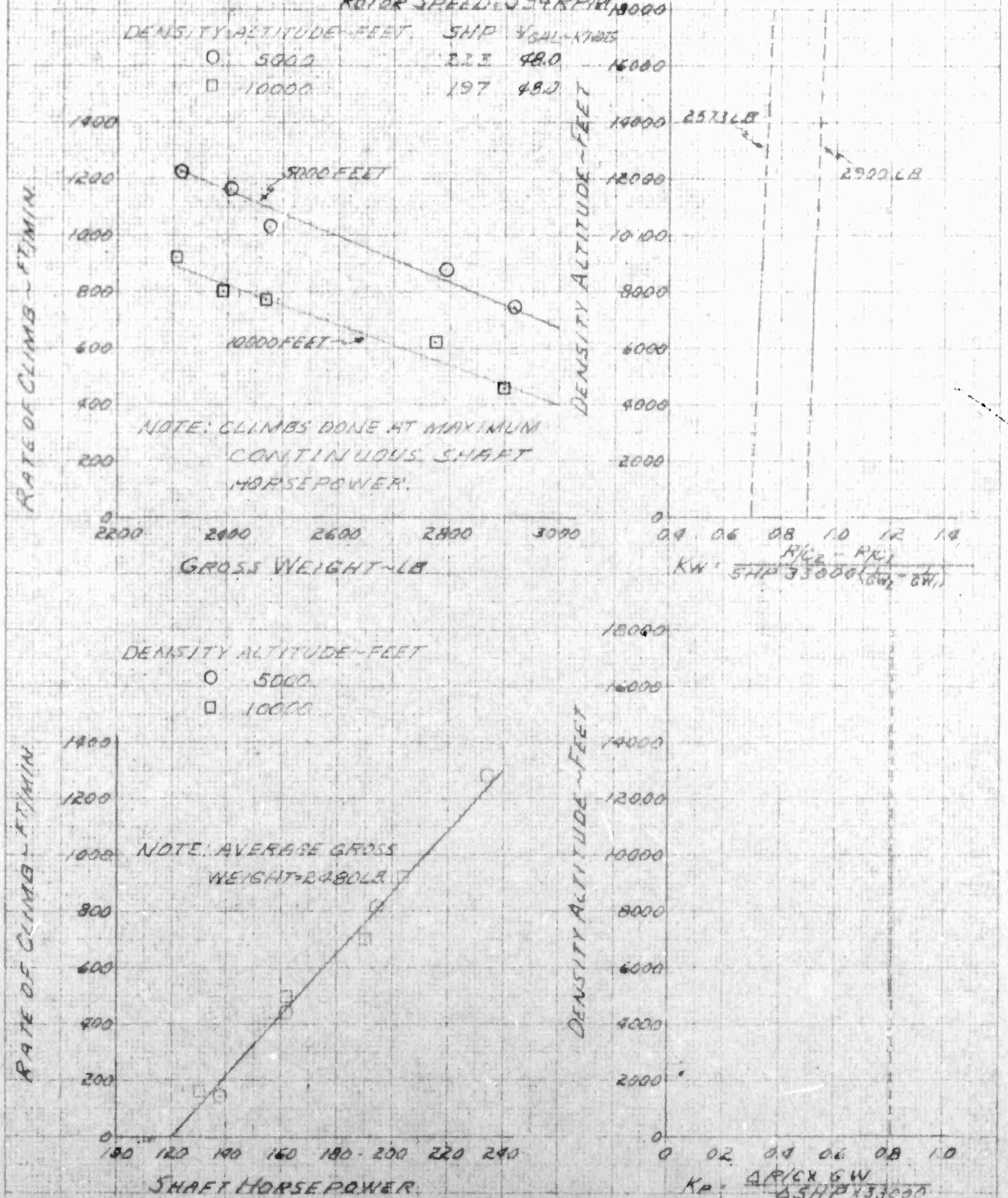


FIGURE No. 34
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
C.G. = STATION 102.0 (MID)
CLEAN CONFIGURATION

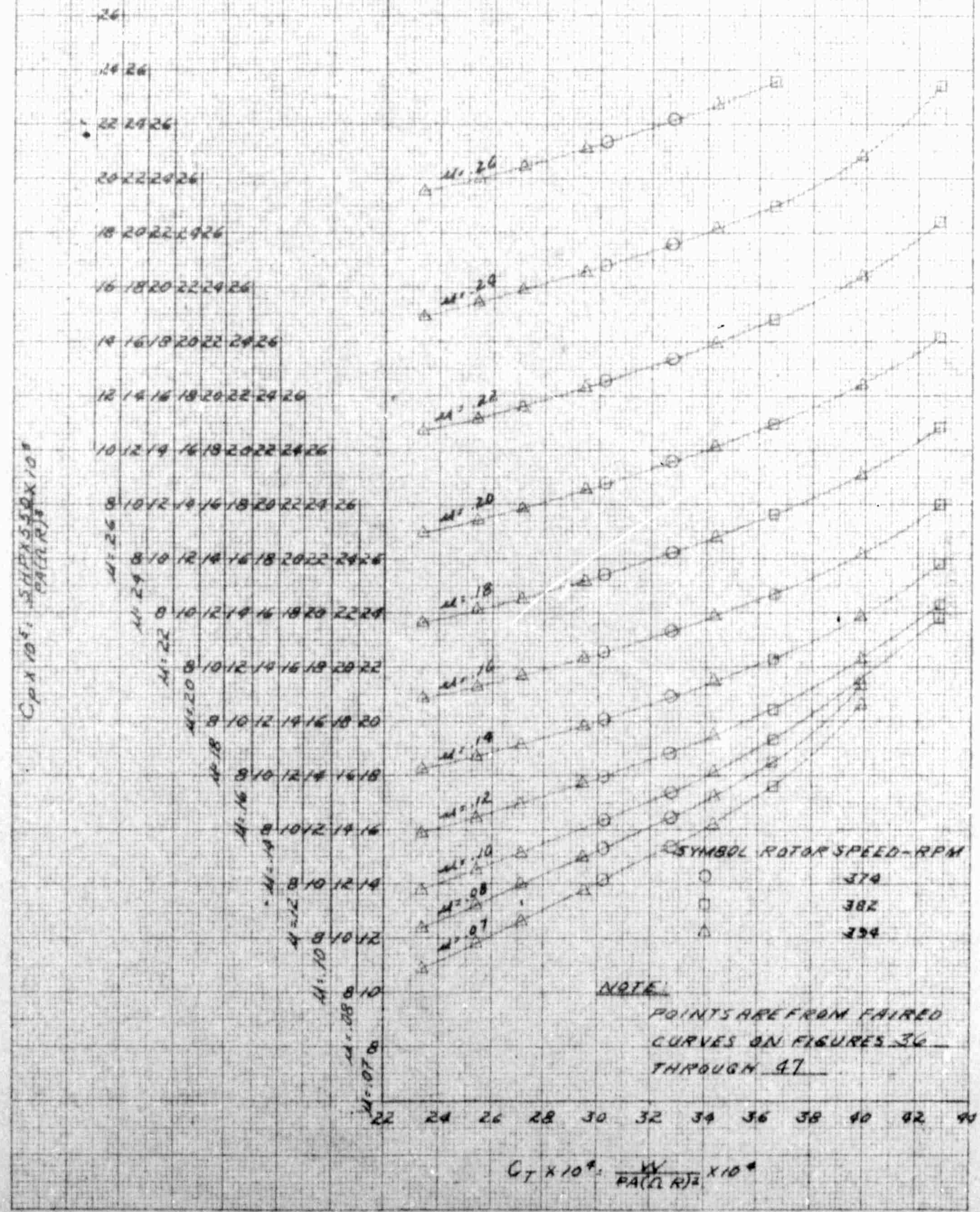


FIGURE NO 35
LEVEL FLIGHT RANGE SUMMARY
OH-4A USA S/N 62-4202

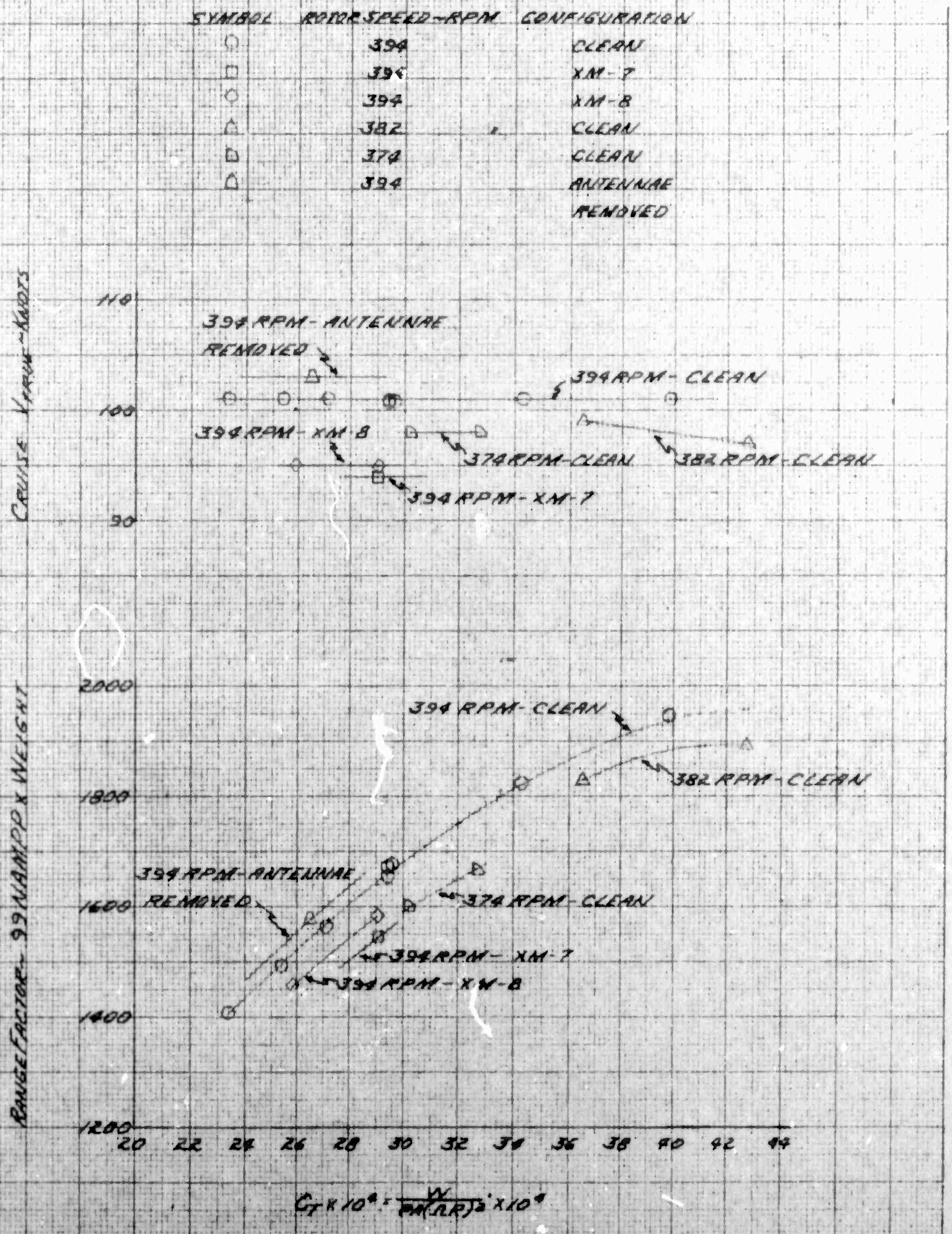
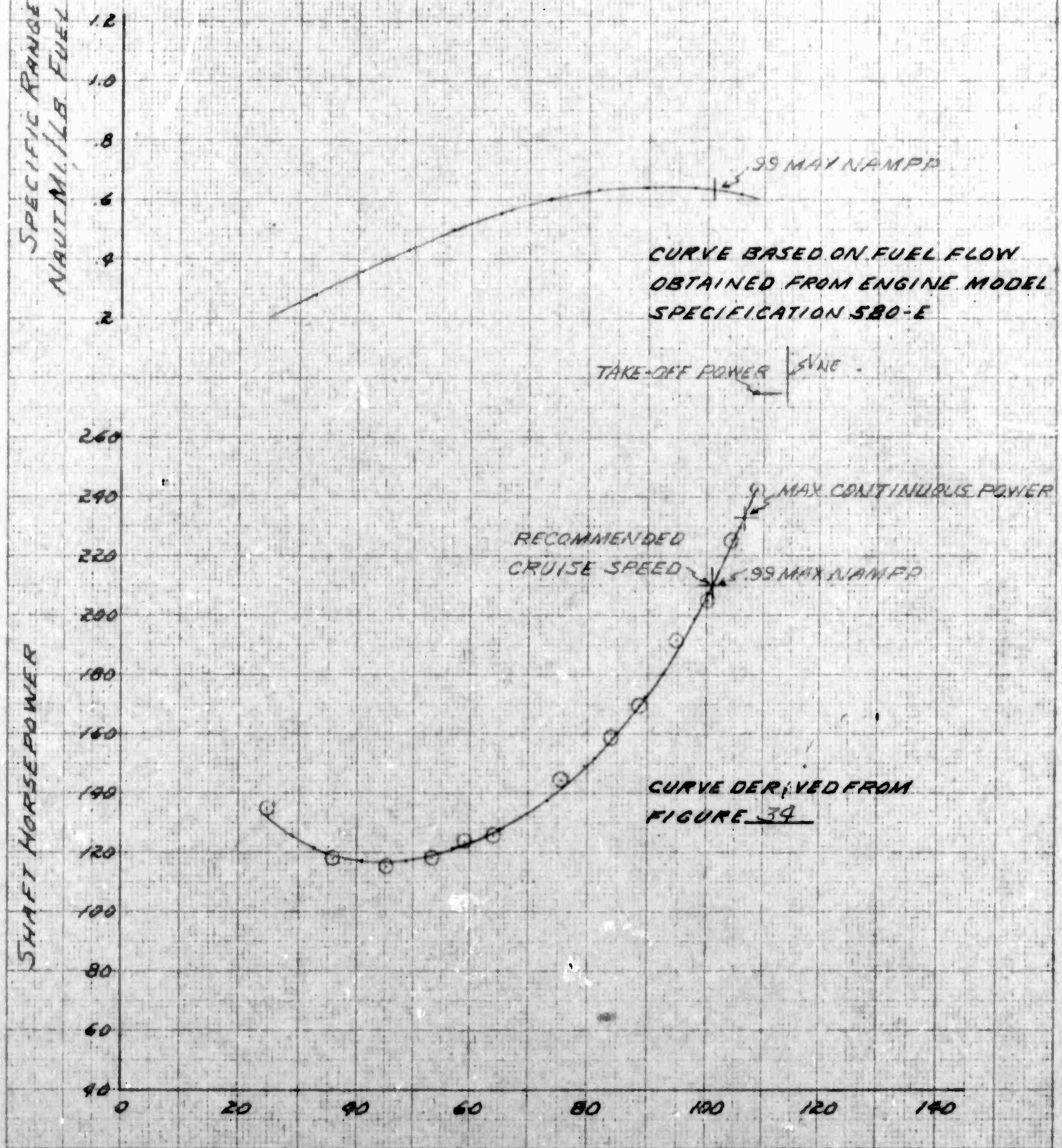


FIGURE NO. 36
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2213 LB
DENSITY ALTITUDE = 1225 FT.
ROTOR SPEED = 394 RPM
C.G. = STATION 102.1 (MID)
$C_T = 23.39 \times 10^{-4}$
CLEAN CONFIGURATION



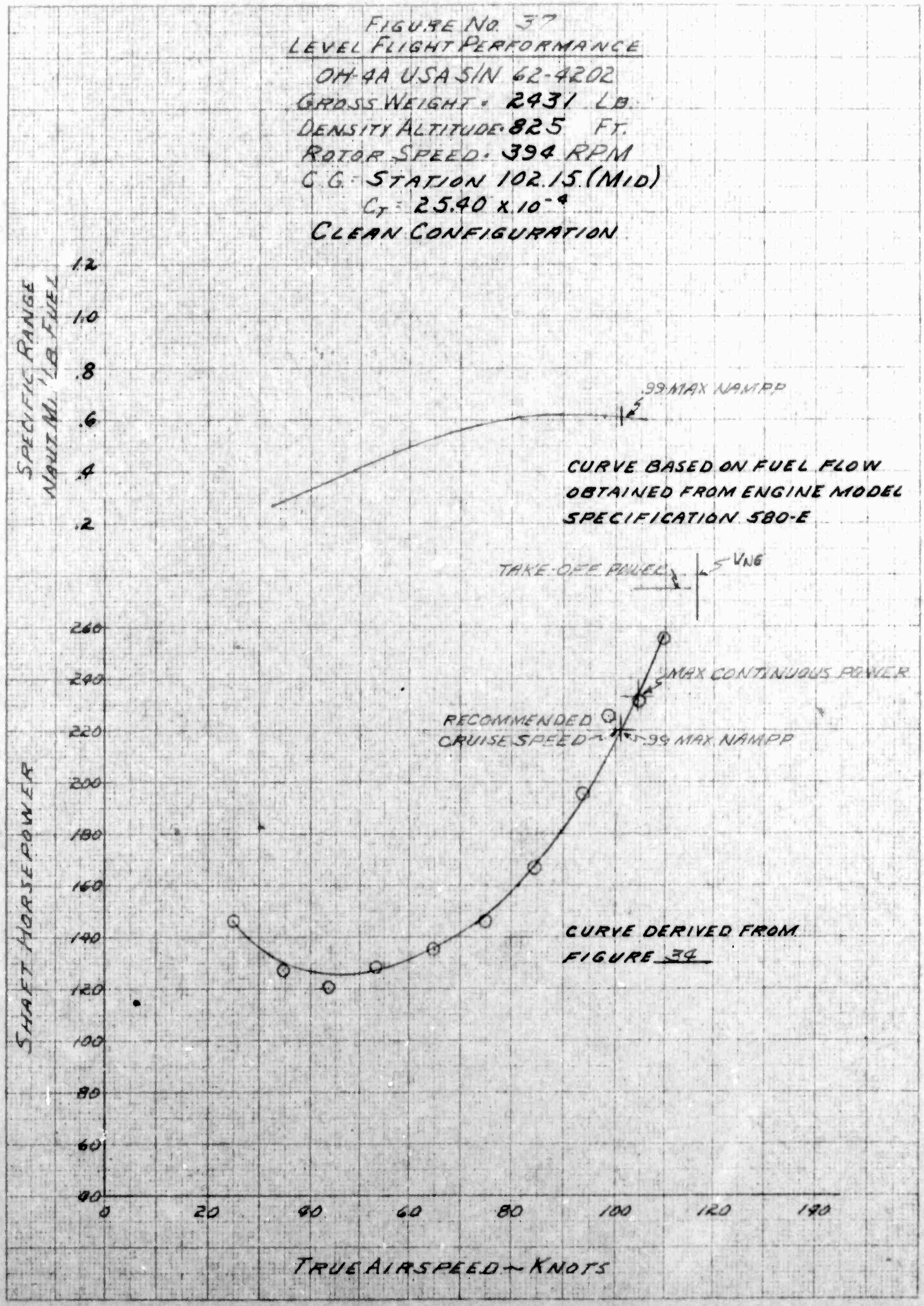


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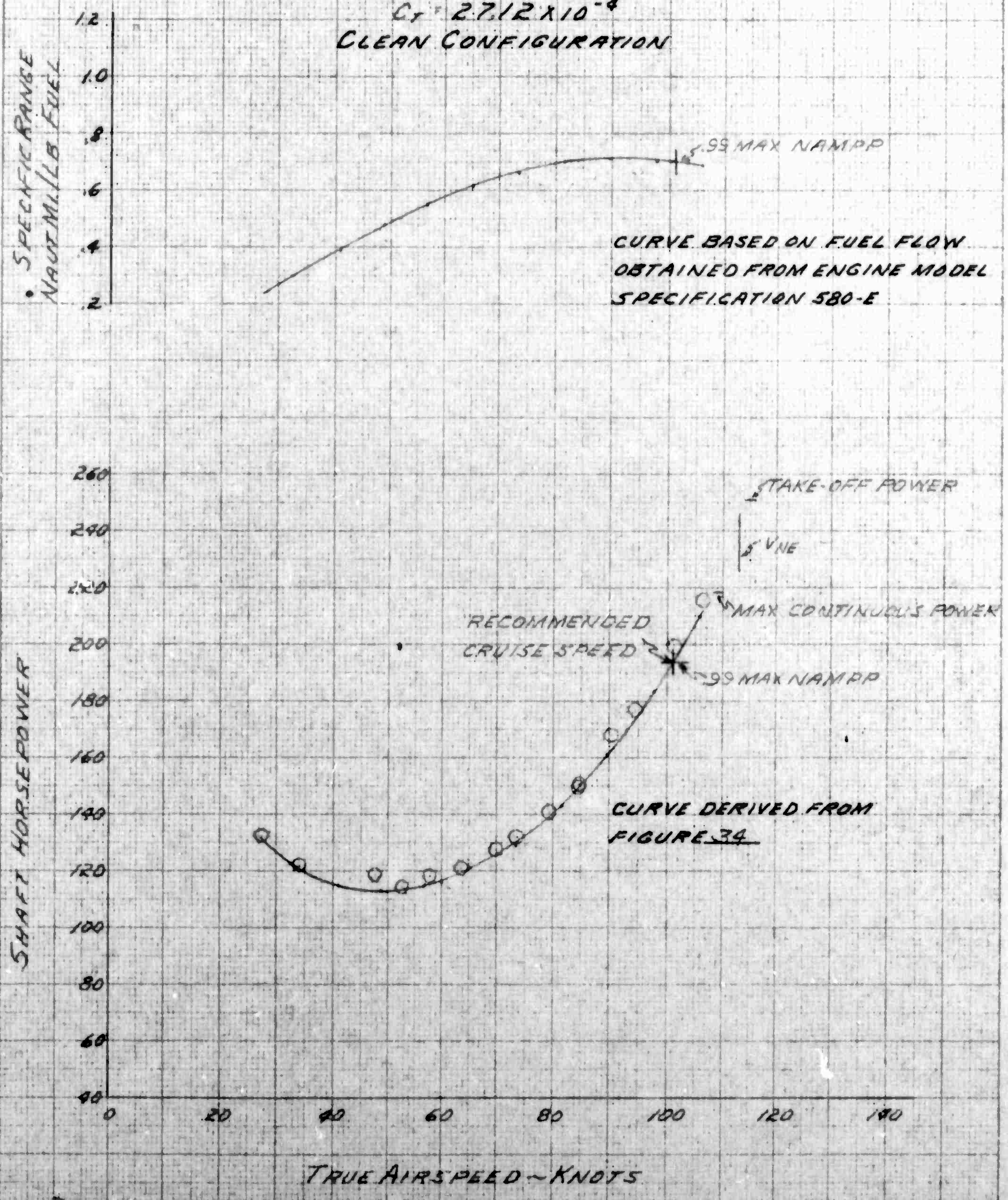
FIGURE NO. 38
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT - 2232 LB.
DENSITY ALTITUDE - 5870 FT.
ROTOR SPEED - 394 RPM
C.G. - STATION 102.2 (MID)
C_T = 27.12 X 10^{-4}
CLEAN CONFIGURATION



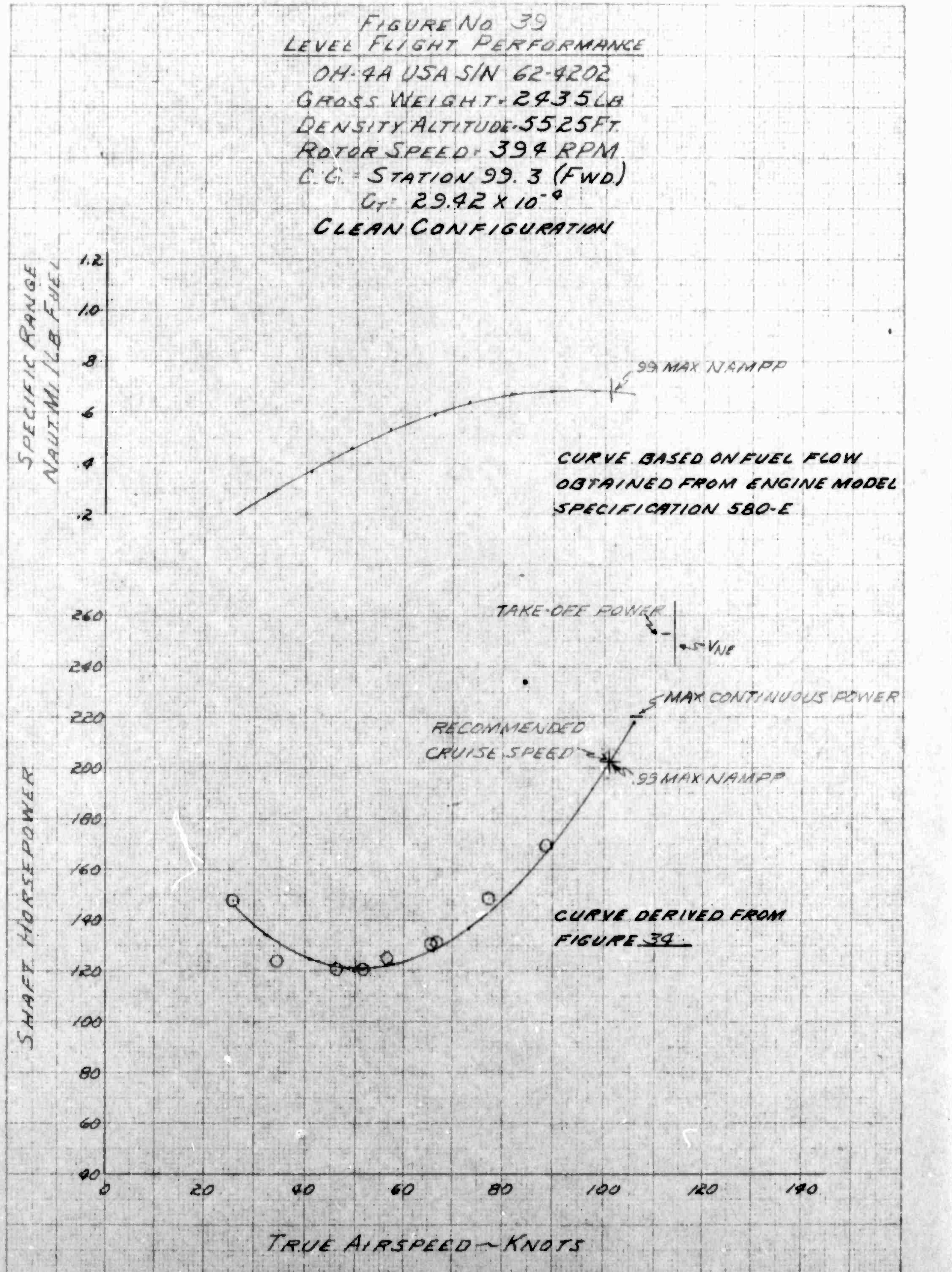


FIGURE NO. 39
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT - 2435 LB
DENSITY ALTITUDE - 5525 FT.
ROTOR SPEED - 394 RPM
C.G. = STATION 99.3 (FWD.)
C_T = 29.42 X 10^{-4}
CLEAN CONFIGURATION
SPECIFIC RANGE NAUT. MI./LB. FUEL
1.2
1.0
.8
.6
.4
.2
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
TAKE-OFF POWER
V_{NE}
MAX CONTINUOUS POWER
RECOMMENDED CRUISE SPEED
.99 MAX NAMPP
CURVE DERIVED FROM FIGURE 34.
SHAFT HORSEPOWER
260
240
220
200
180
160
140
120
100
80
60
40
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

FIGURE NO. 40
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2389 LB.
DENSITY ALTITUDE = 6350 FT
ROTOR SPEED = 394 RPM
C.G. = STATION 106.3 (AFT)
$C_T = 29.45 \times 10^{-4}$
CLEAN CONFIGURATION

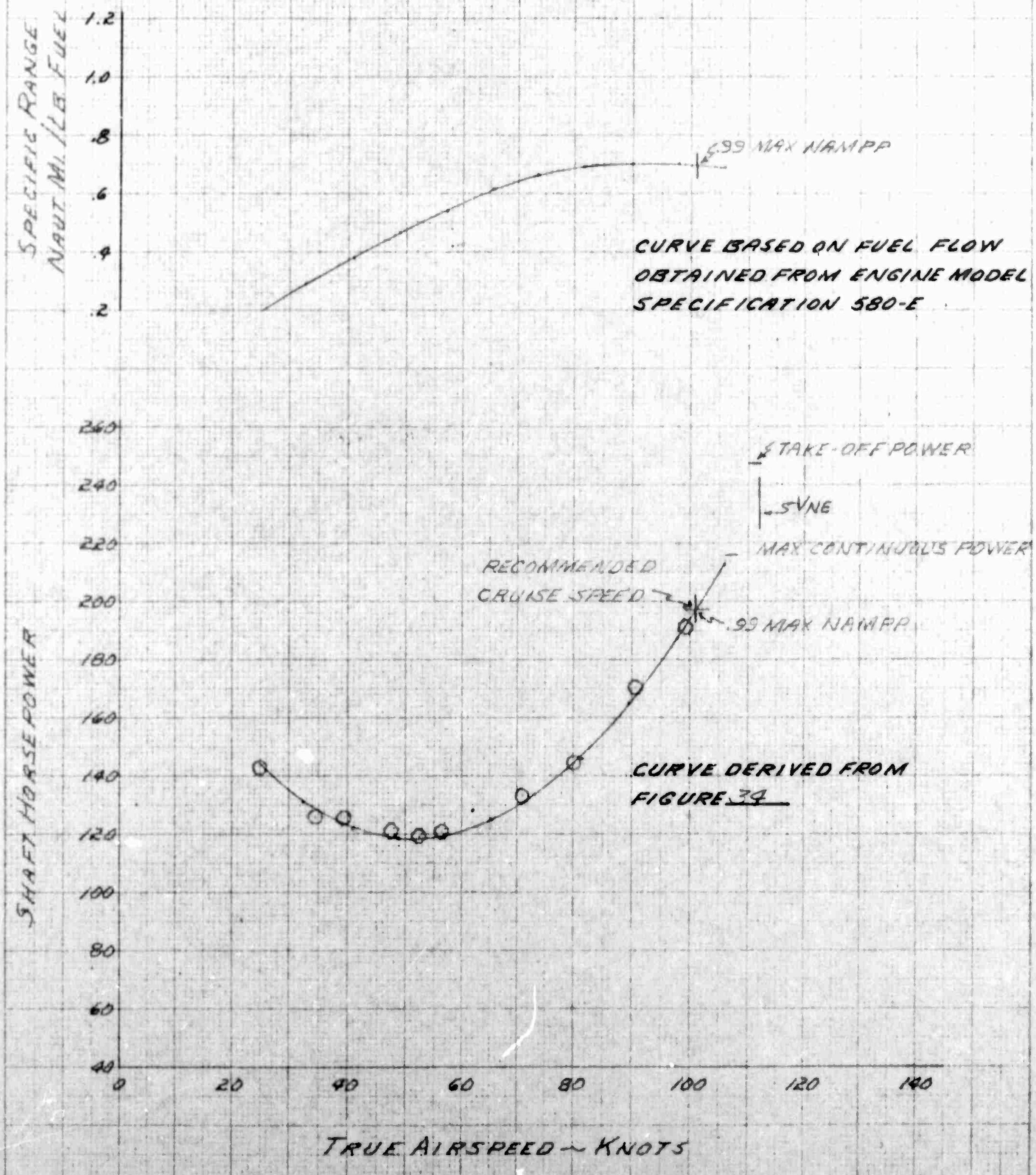


FIGURE NO. 41
LEVEL FLIGHT PERFORMANCE

OH-4A USA S/N 62-4202
GROSS WEIGHT 2499 LB.
DENSITY ALTITUDE 4905 FT
ROTOR SPEED 394 RPM
C.G. STATION 102.4 (MID)
$C_T = 2952 \times 10^{-4}$
CLEAN CONFIGURATION

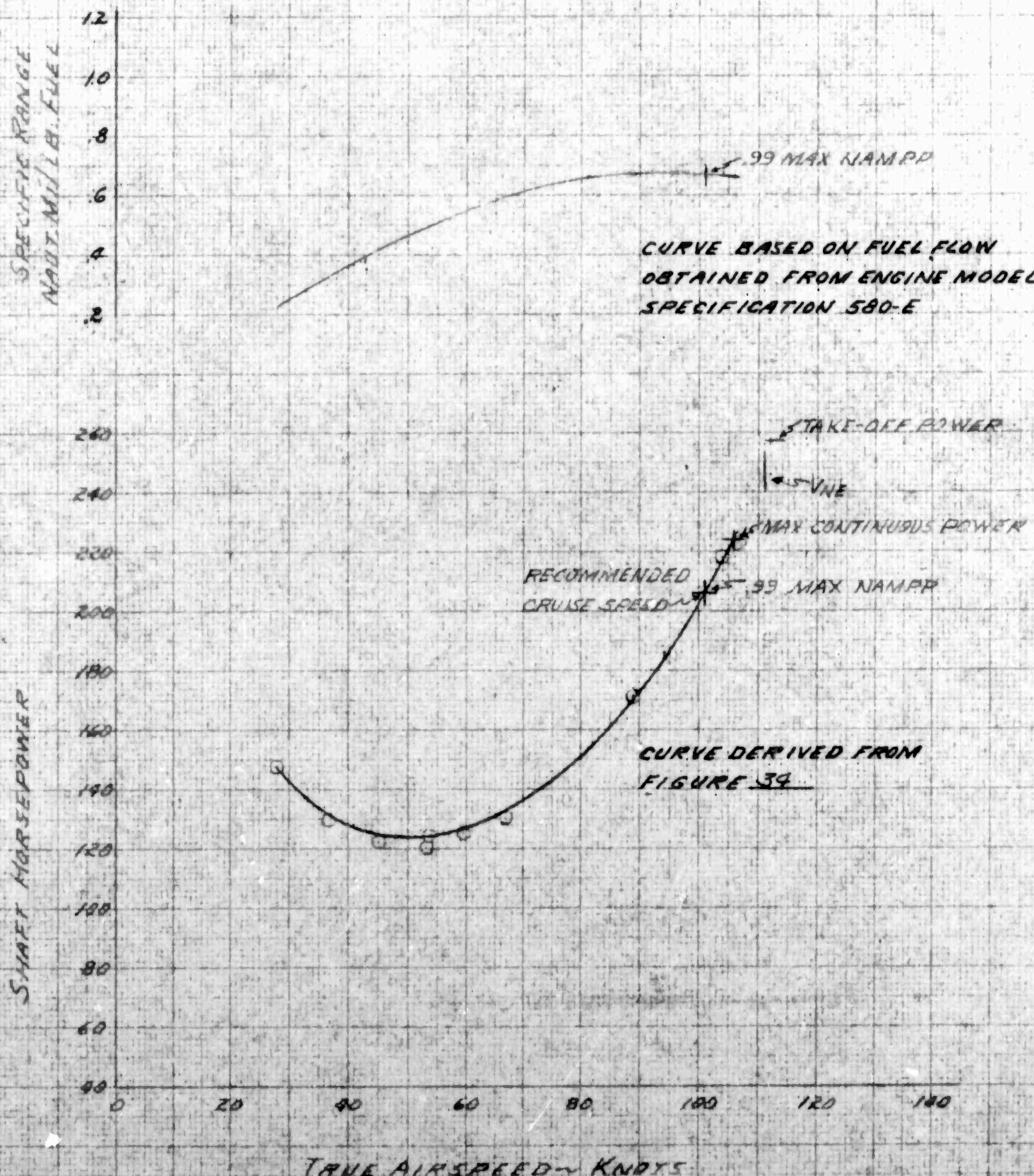


Figure No. 42
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT: 2245 LB
DENSITY ALTITUDE: 5805 FT.
ROTOR SPEED: 374 RPM
C.G.: STATION 102.3 (MID)
C_T: 3022×10^{-4}
CLEAN CONFIGURATION

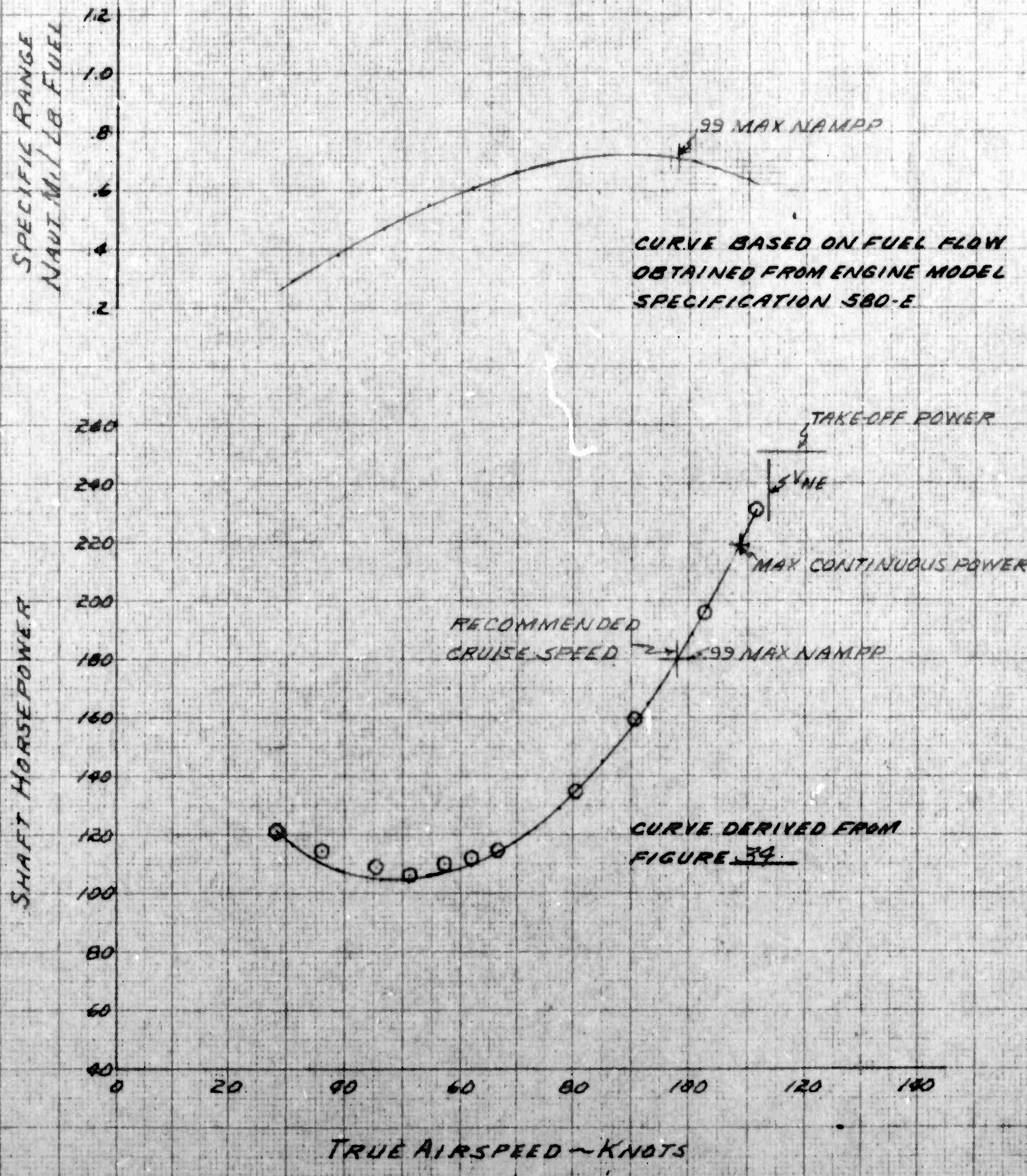


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FIGURE NO. 43
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2806 LB.
DENSITY ALTITUDE = 1030 FT.
ROTOR SPEED = 374 RPM
C.G. = STATION 101.8 (MID)
$C_T = 32.71 \times 10^{-4}$
CLEAN CONFIGURATION

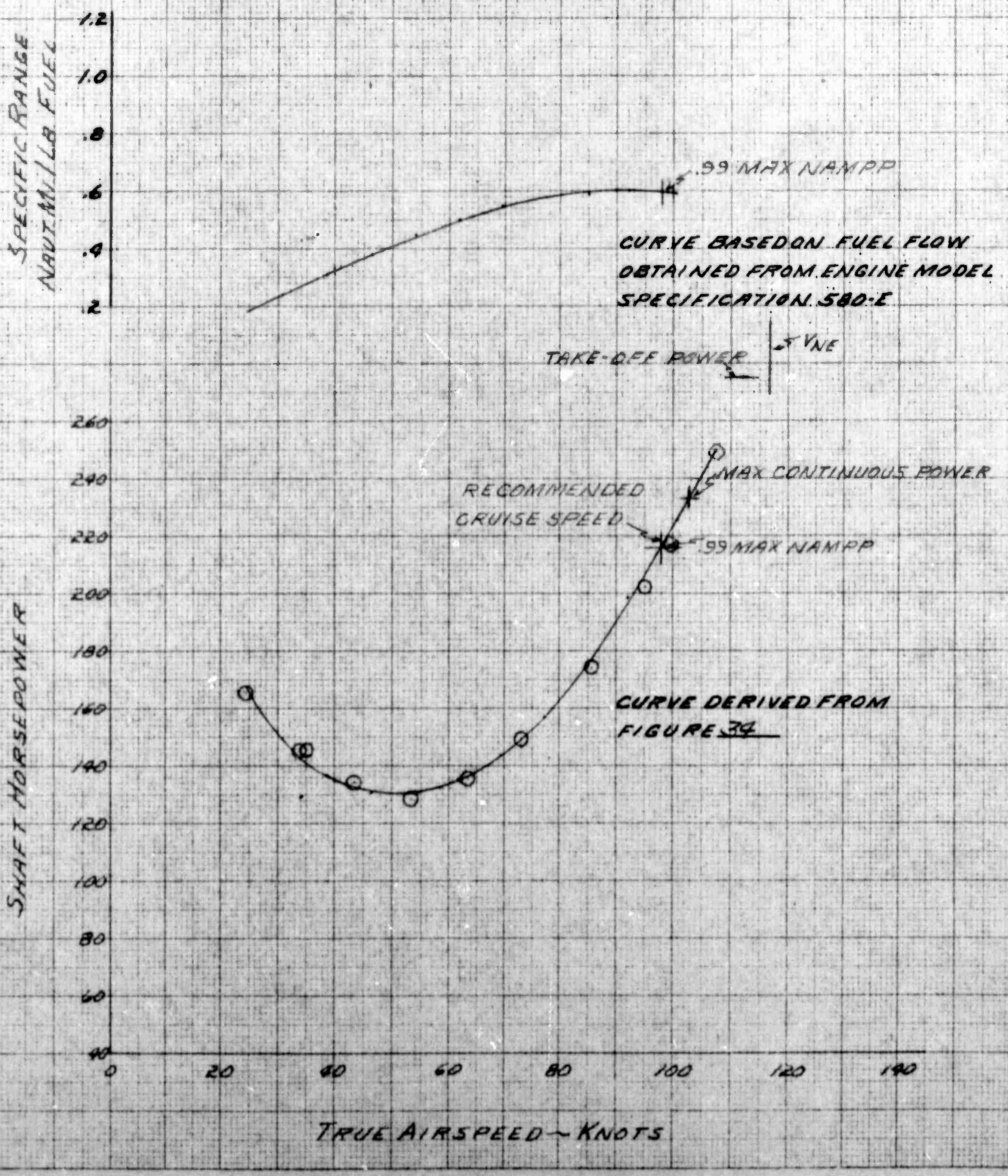

FIGURE NO. 44
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2460 LB.
DENSITY ALTITUDE = 10370 FT.
ROTOR SPEED = 394 RPM
C.G. = STATION 102.2 (MID)
$C_T = 34.30 \times 10^{-4}$
CLEAN CONFIGURATION

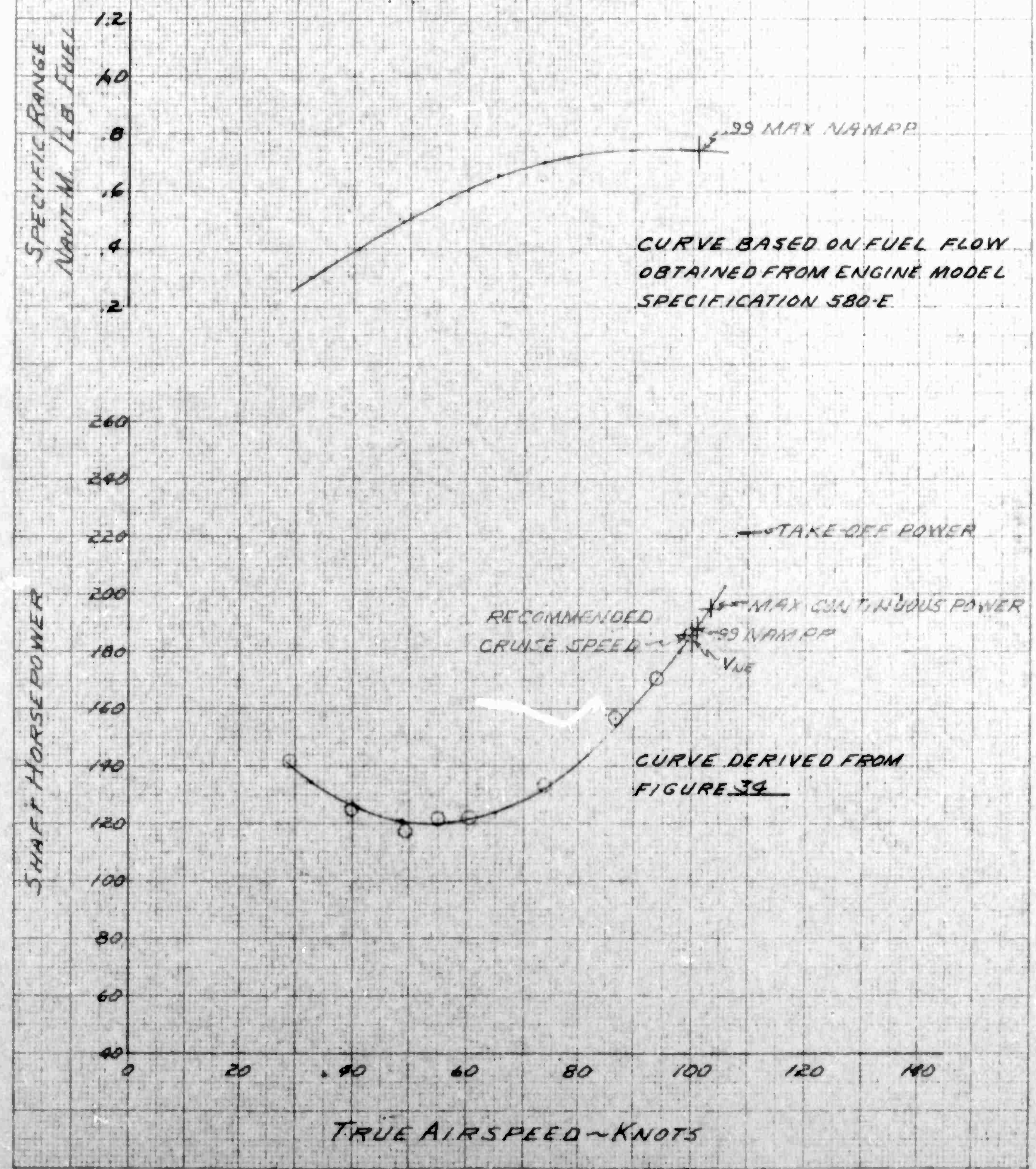

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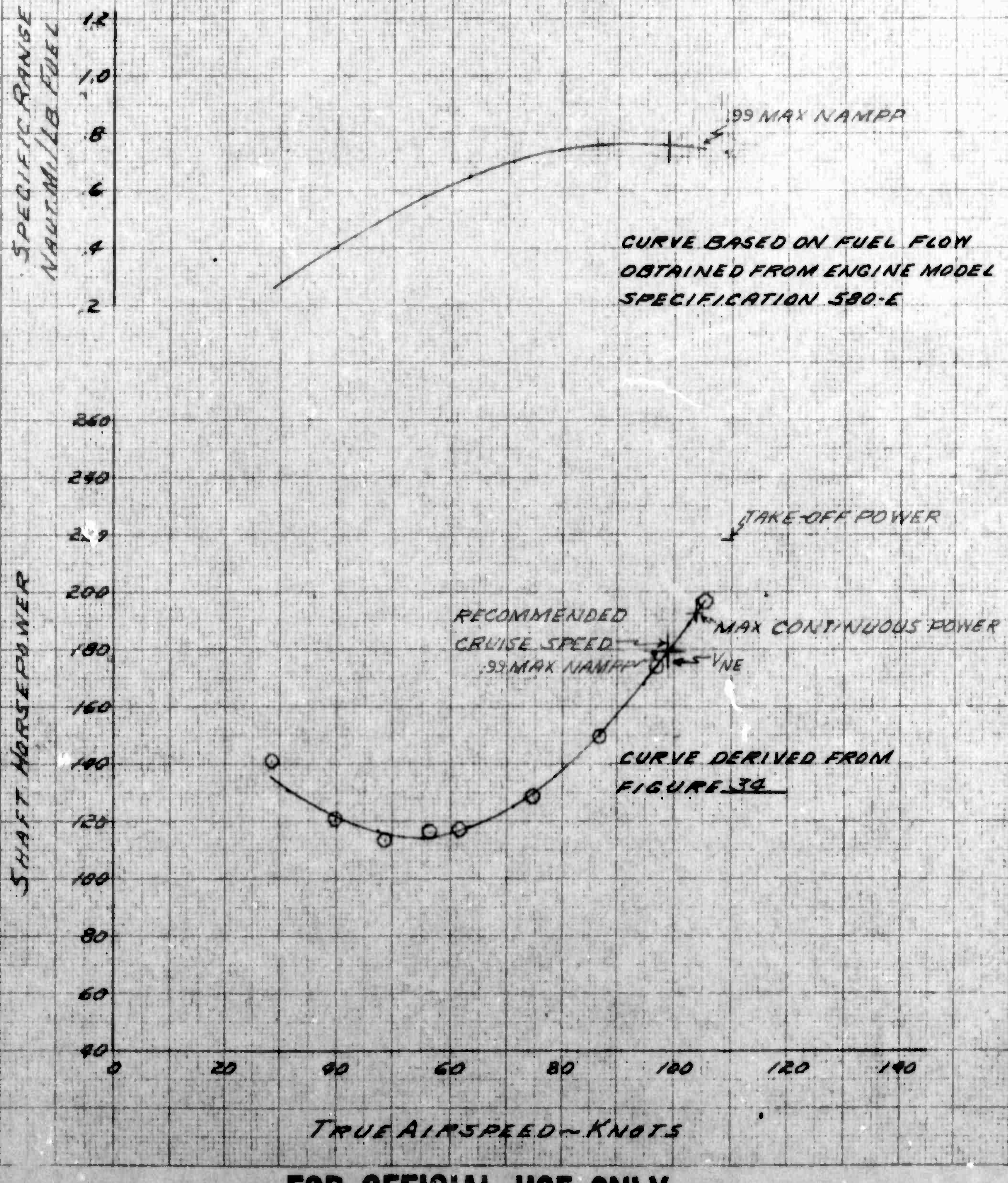
FIGURE NO. 45
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2435 LB
DENSITY ALTITUDE = 10730 FT
ROTOR SPEED = 382 RPM
C.G. = STATION 101.2 (MID)
$C_T = 36.48 \times 10^{-4}$
CLEAN CONFIGURATION



FOR OFFICIAL USE ONLY

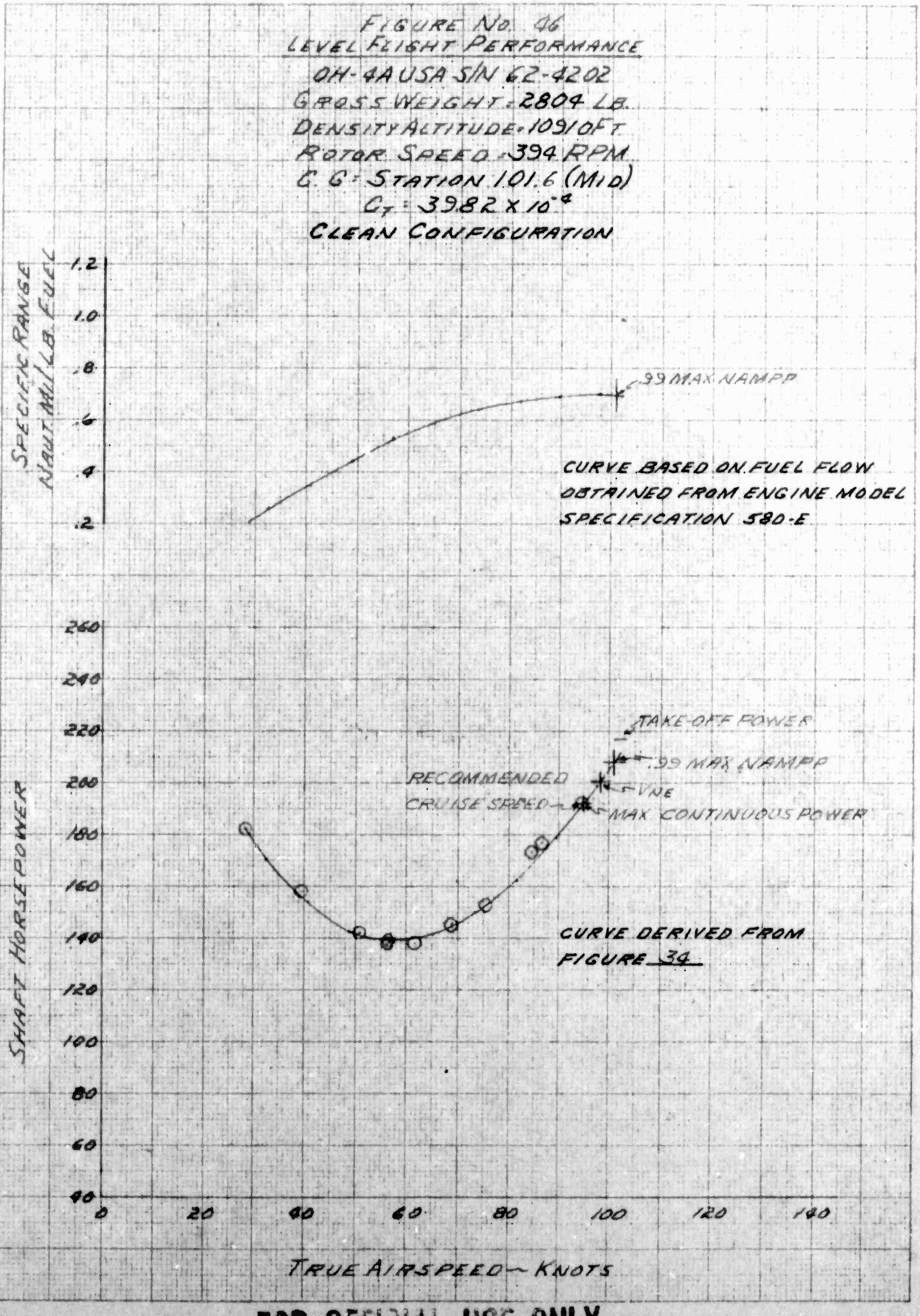


FIGURE NO. 46
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2804 LB.
DENSITY ALTITUDE = 10910 FT
ROTOR SPEED = 394 RPM
C.G. = STATION 101.6 (MID)
C_T = 39.82 x 10^{-4}
CLEAN CONFIGURATION
SPECIFIC RANGE NAUT. MI./LB. FUEL
1.2
1.0
.8
.6
.4
.2
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
SHAFT HORSEPOWER
260
240
220
200
180
160
140
120
100
80
60
40
TAKE-OFF POWER
.99 MAX NAMPP
RECOMMENDED CRUISE SPEED
VNE
MAX CONTINUOUS POWER
CURVE DERIVED FROM FIGURE 34
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

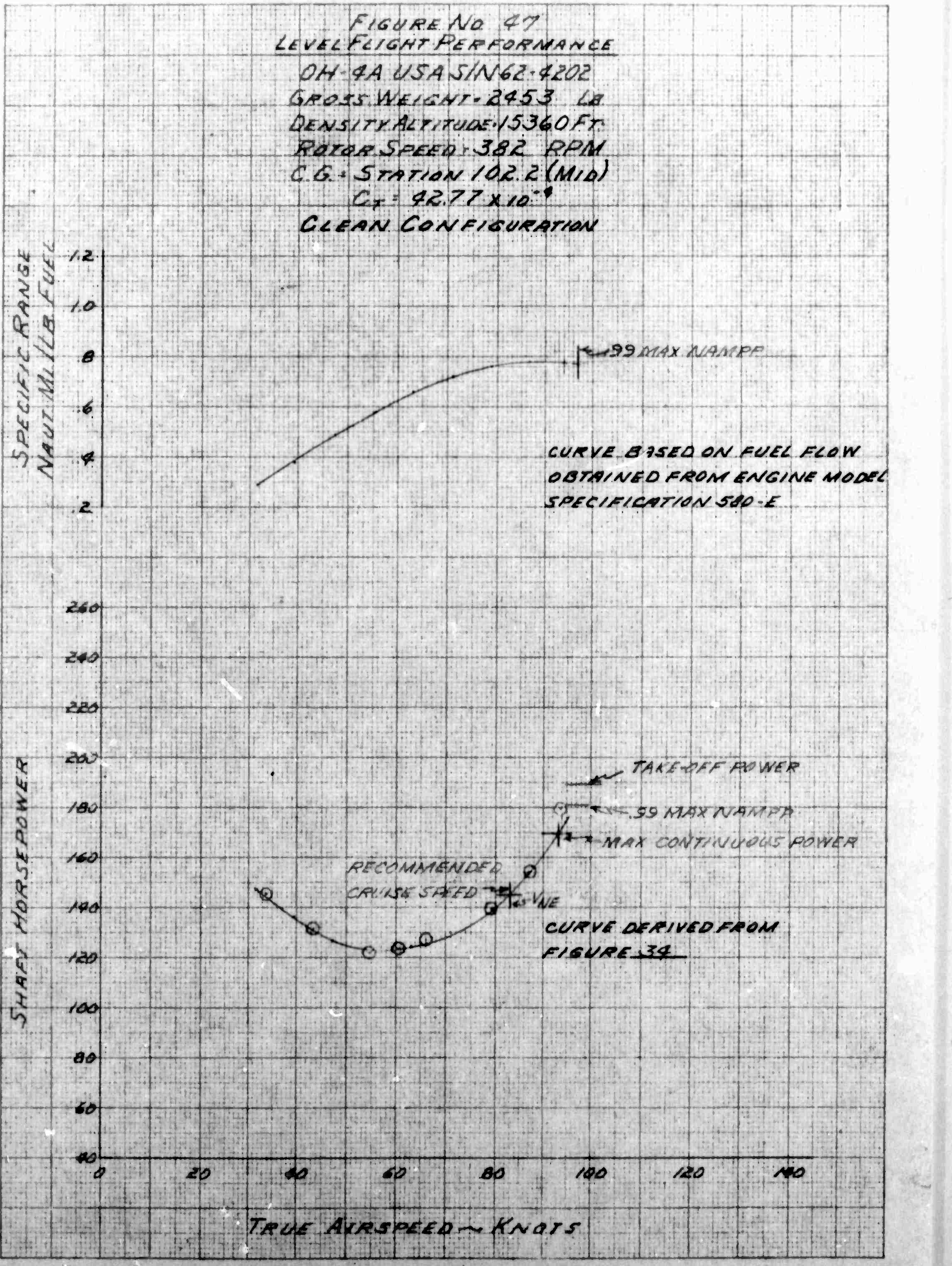


Figure No. 47
Level Flight Performance
OH-4A USA S/N 62-4202
Gross Weight = 2453 Lb
Density Altitude = 15360 Ft
Rotor Speed = 382 RPM
C.G. = Station 102.2 (Mid)
$C_T = 42.77 \times 10^{-4}$
Clean Configuration

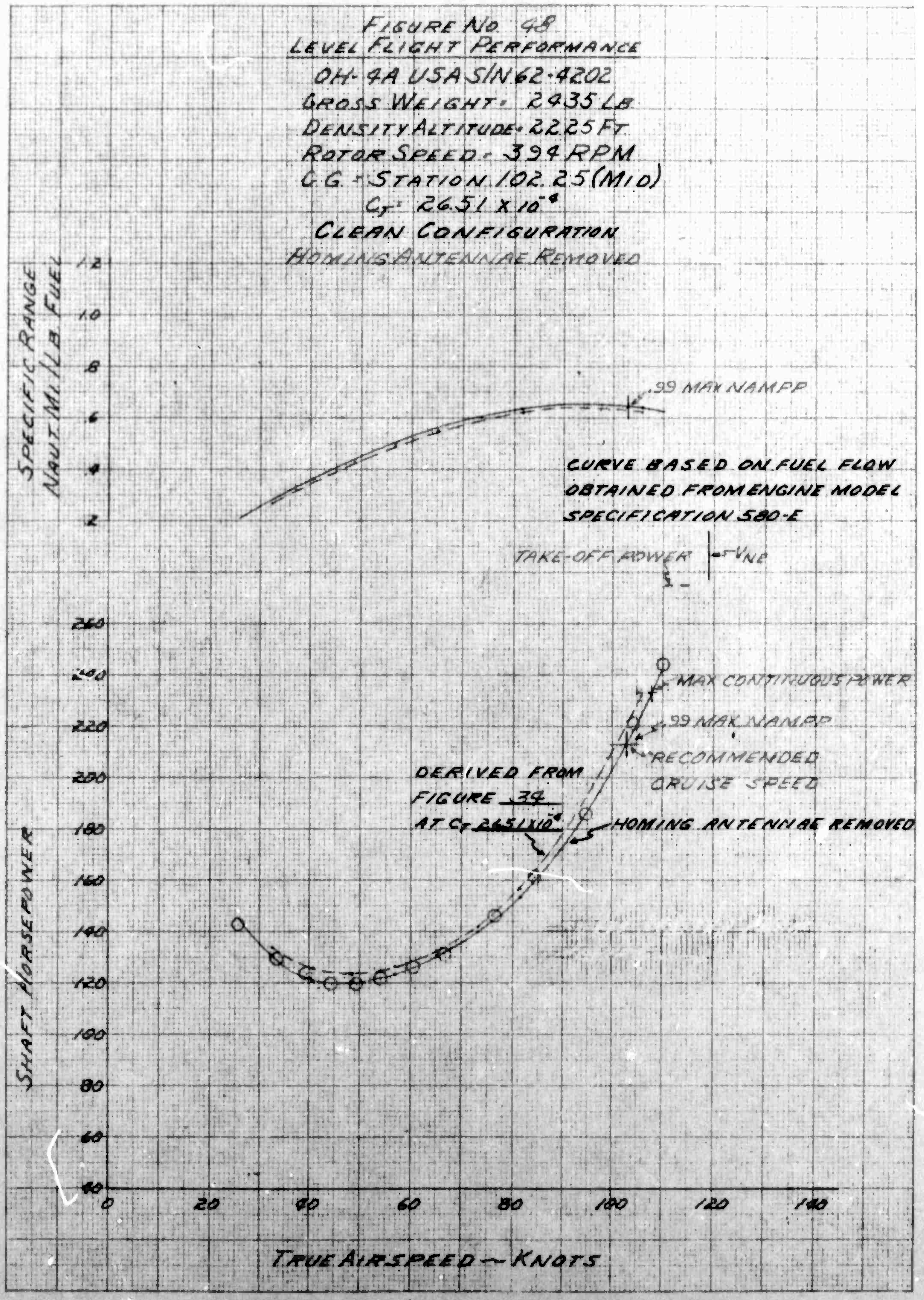
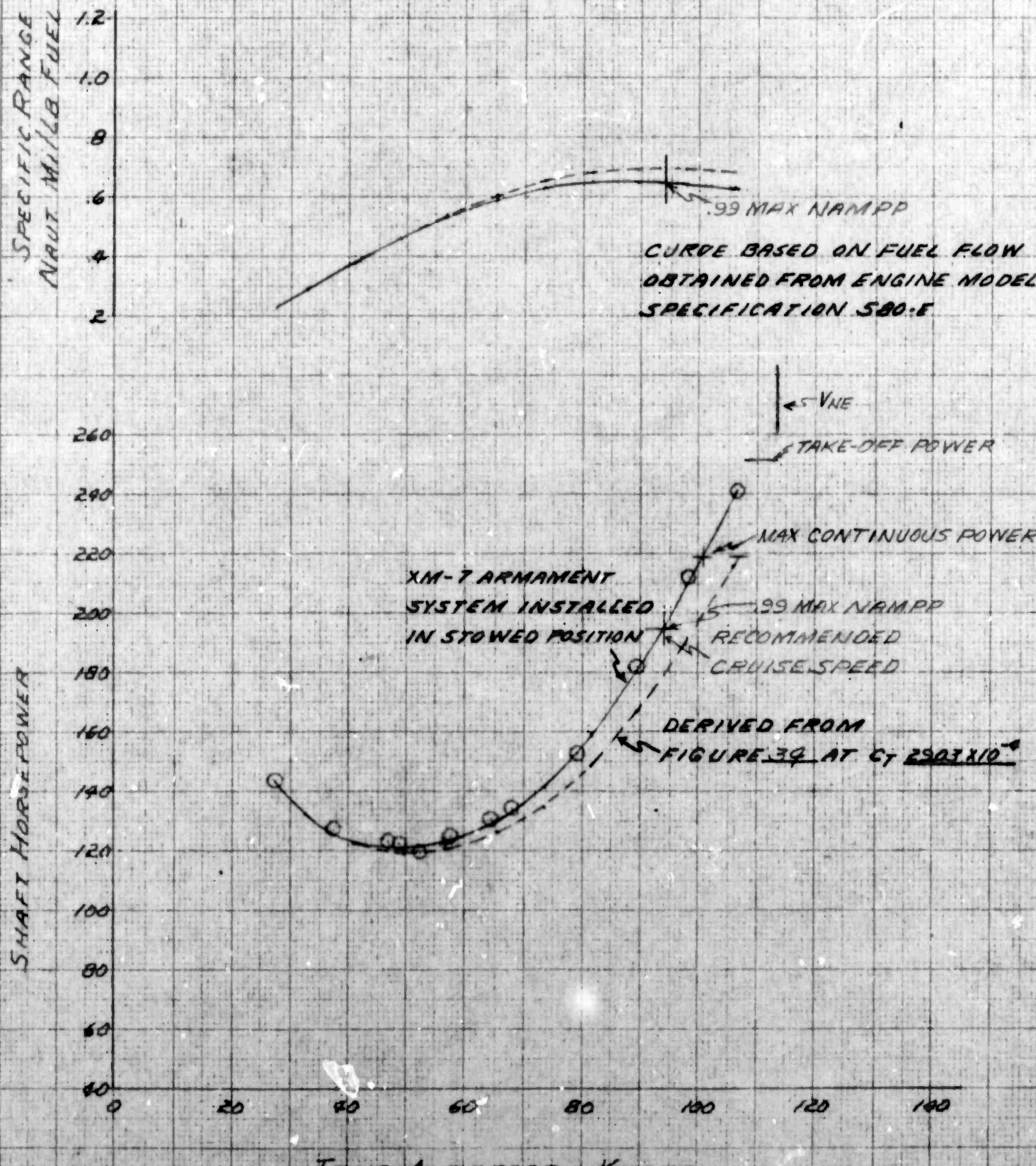
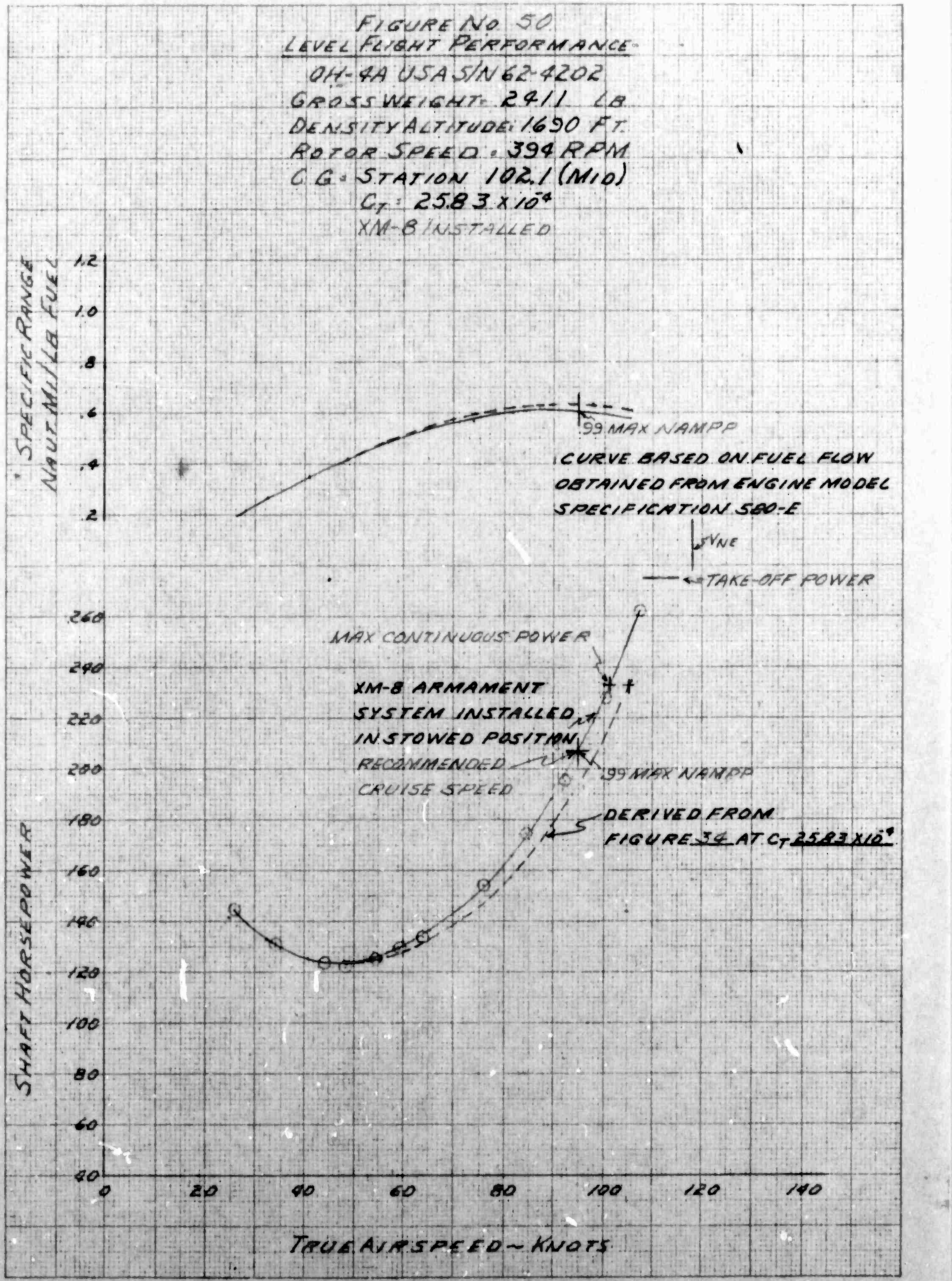


FIGURE No. 48
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2435 LB
DENSITY ALTITUDE = 2225 FT
ROTOR SPEED = 394 RPM
C.G. = STATION 102.25 (MID)
$C_T = 26.51 \times 10^{-4}$
CLEAN CONFIGURATION
HOMING ANTENNAE REMOVED
SPECIFIC RANGE
NAUT MI/LB FUEL
1.2
1.0
.8
.6
.4
.2
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW OBTAINED FROM ENGINE MODEL SPECIFICATION 580-E
TAKE-OFF POWER
V_{NE}
SHAFT HORSEPOWER
260
240
220
200
180
160
140
120
100
80
60
40
MAX CONTINUOUS POWER
.99 MAX NAMPP
RECOMMENDED CRUISE SPEED
DERIVED FROM FIGURE 39 AT C_T 26.51×10^{-4}
HOMING ANTENNAE REMOVED
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

FIGURE NO. 49
LEVEL FLIGHT PERFORMANCE

OH-4A USA S/N 62-4202
GROSS WEIGHT = 2397 LB
DENSITY ALTITUDE = 5765 FT
ROTOR SPEED = 394 RPM
C.G. = STATION 102.4 (MID)
$C_T = 2903 \times 10^{-4}$
XM-7 INSTALLED





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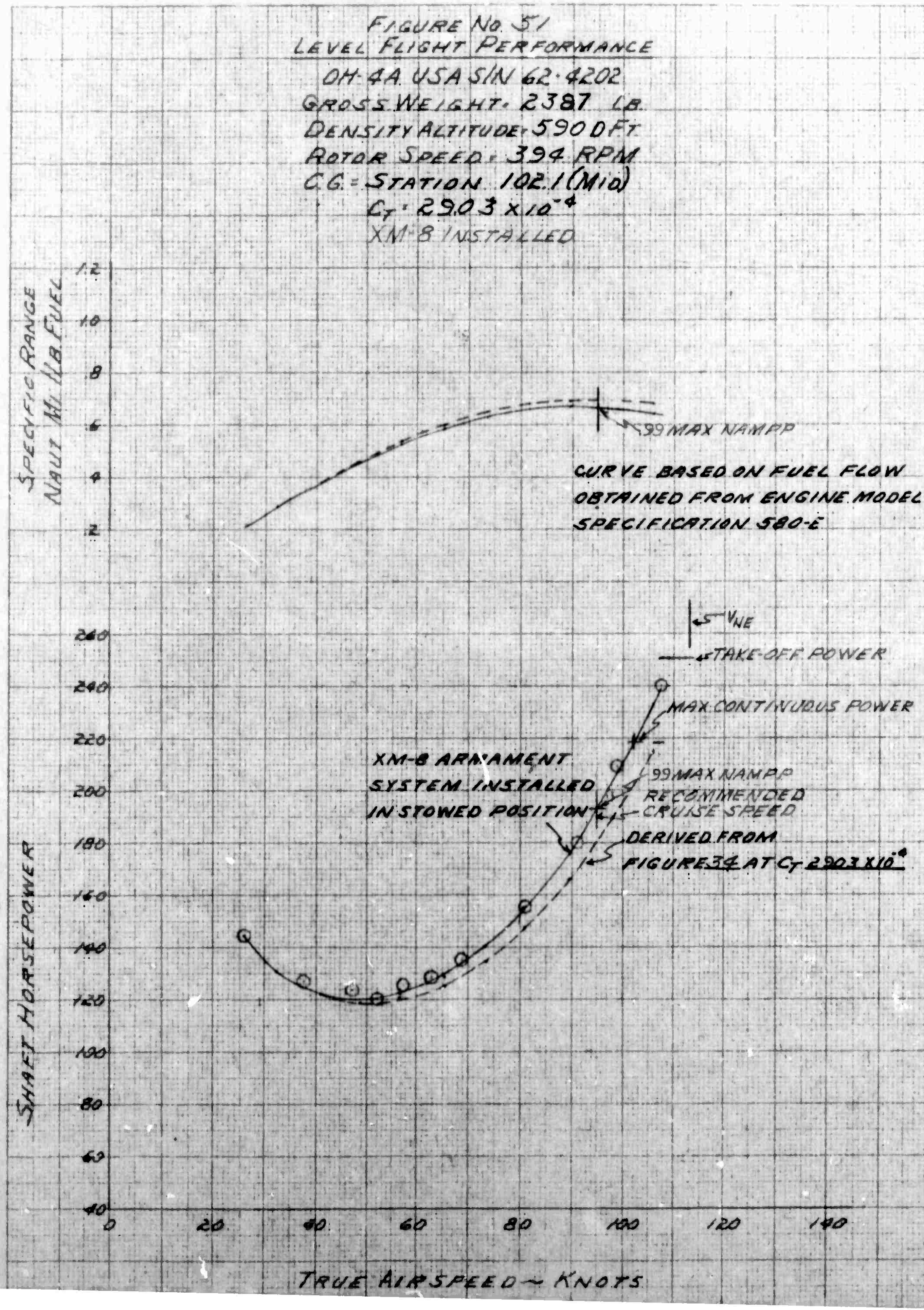


FIGURE NO. 51
LEVEL FLIGHT PERFORMANCE
OH-4A USA S/N 62-4202
GROSS WEIGHT = 2387 LB.
DENSITY ALTITUDE = 5900 FT.
ROTOR SPEED = 394 RPM
C.G. = STATION 102.1 (MID)
C_T = 29.03 x 10^{-4}
XM-8 INSTALLED
SPECIFIC RANGE
NAUT MI/LB FUEL
.99 MAX NAMPP
CURVE BASED ON FUEL FLOW
OBTAINED FROM ENGINE MODEL
SPECIFICATION 580-E
V_{NE}
TAKE-OFF POWER
MAX CONTINUOUS POWER
XM-8 ARMAMENT
SYSTEM INSTALLED
IN STOWED POSITION
.99 MAX NAMPP
RECOMMENDED
CRUISE SPEED
DERIVED FROM
FIGURE 34 AT C_T = 29.03 x 10^{-4}
SHAFT HORSEPOWER
TRUE AIRSPEED ~ KNOTS

FIGURE NO. 52
AUTOROTATION DESCENTS
OH-4A USA S/N 62-4202
ROTOR SPEED = 394 RPM
C.G. STATION = 101.5 (MID)
CLEAN CONFIGURATION

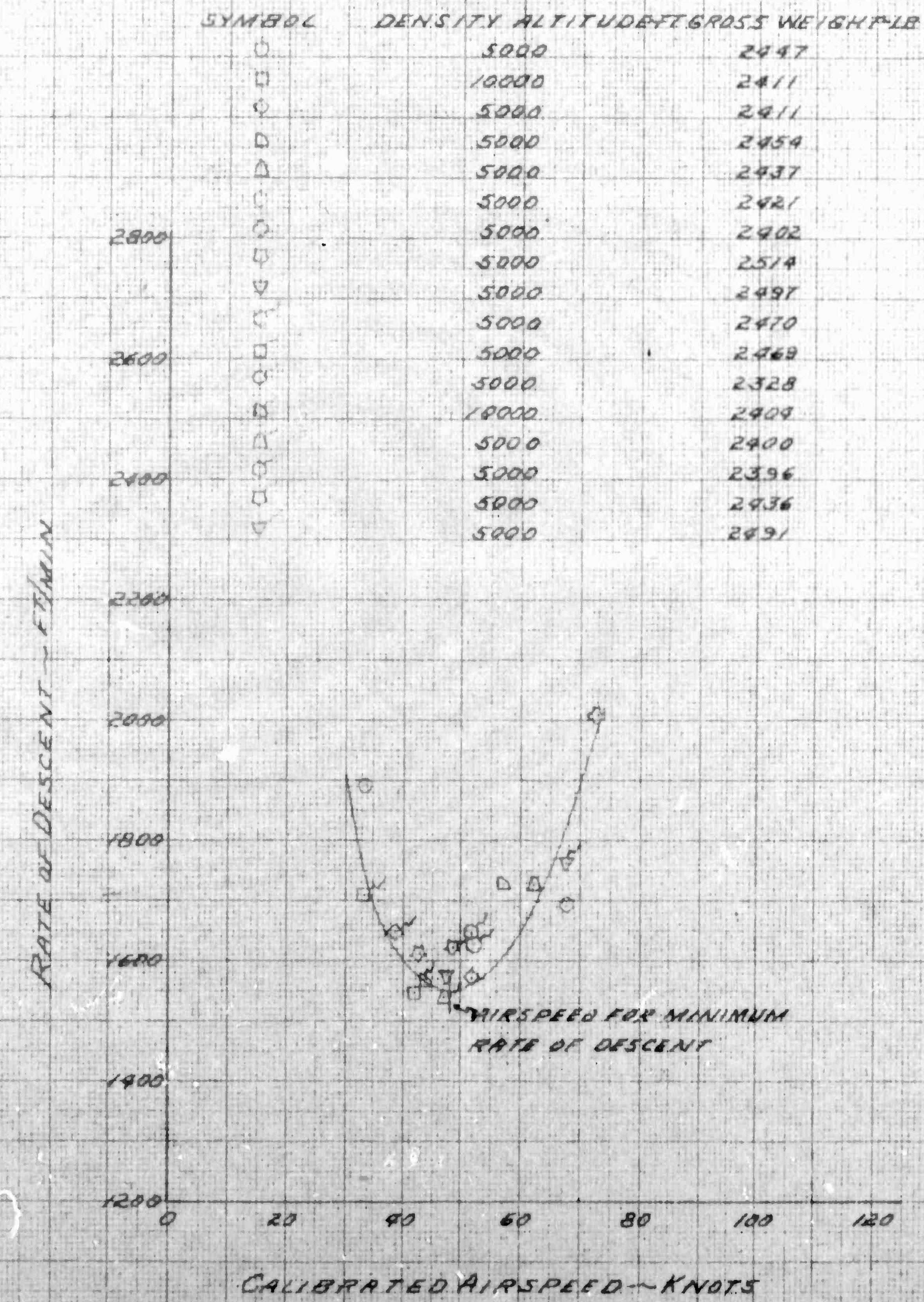
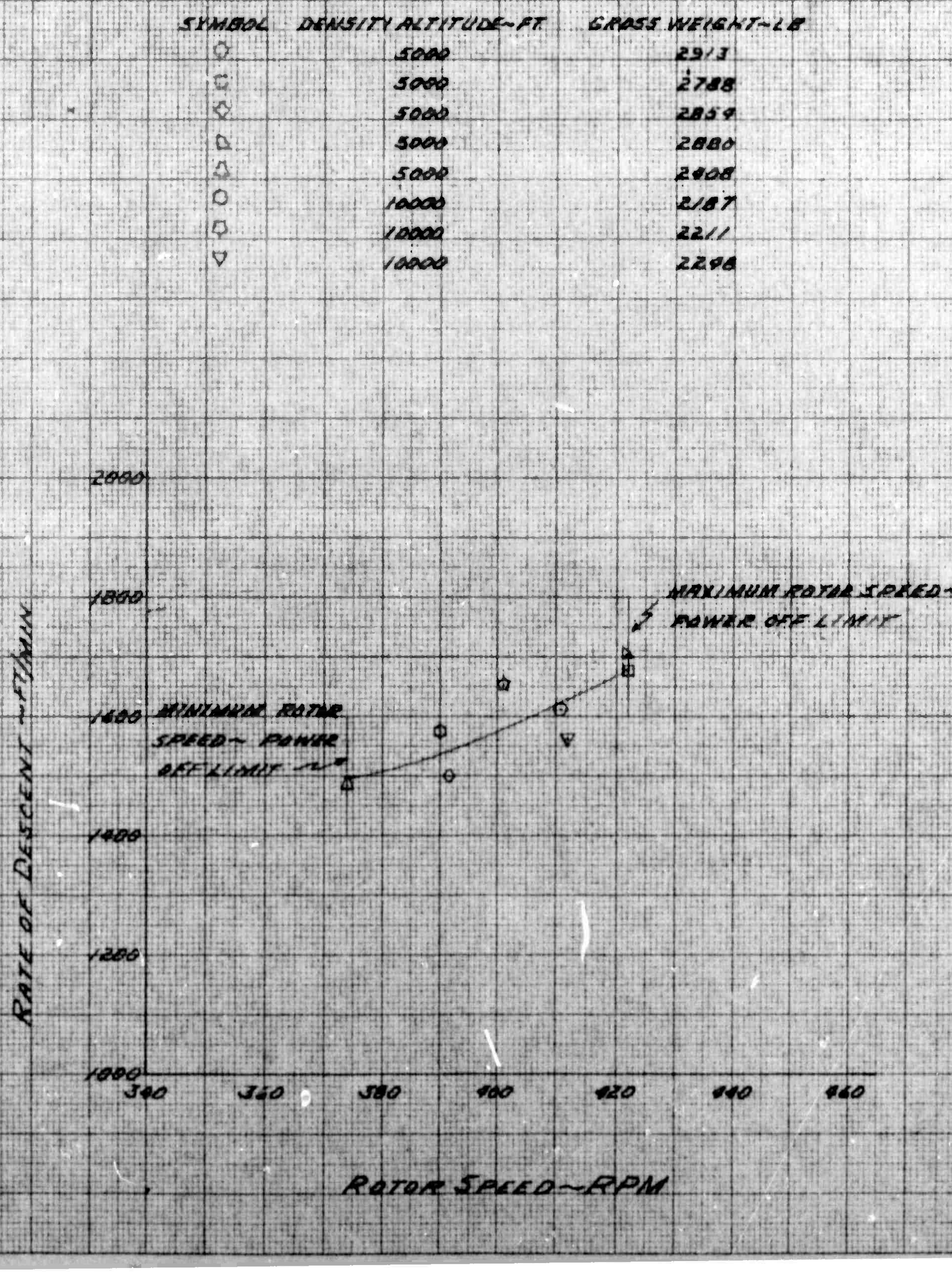


FIGURE NO. 53
AUTOROTATION DESCENTS
OH-4A USA S/N 62-4202
AIRSPEED: 48.5 KNOTS CAS
C.G.: STATION 101.5 (MID)
CLEAN CONFIGURATION



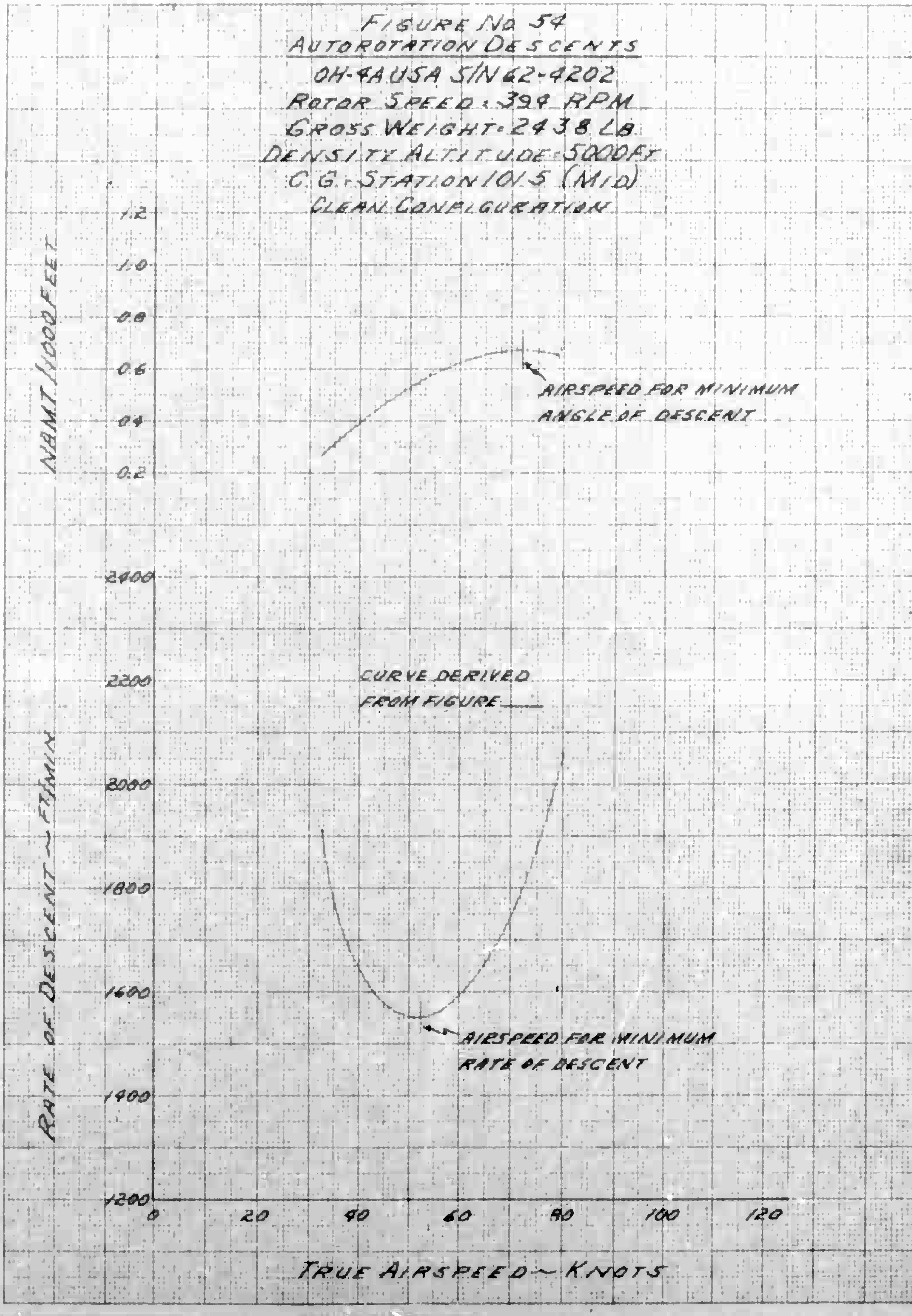


FIGURE NO. 54
AUTOROTATION DESCENTS
OH-4A USA S/N 62-4202
ROTOR SPEED: 394 RPM
GROSS WEIGHT: 2438 LB
DENSITY ALTITUDE: 5000 FT
C.G.: STATION 101.5 (MID)
CLEAN CONFIGURATION
NAMT/1000 FEET
1.2
1.0
0.8
0.6
0.4
0.2
AIRSPEED FOR MINIMUM ANGLE OF DESCENT
RATE OF DESCENT ~ FT/MIN
2400
2200
2000
1800
1600
1400
1200
CURVE DERIVED FROM FIGURE ___
AIRSPEED FOR MINIMUM RATE OF DESCENT
0
20
40
60
80
100
120
TRUE AIRSPEED ~ KNOTS

FIGURE NO 55

SHAFT HORSEPOWER AVAILABLE

OH-4A USA S/N 62-4202

T63-A-5 ENGINE

TAKE-OFF POWER

(738 °C T_{T5})

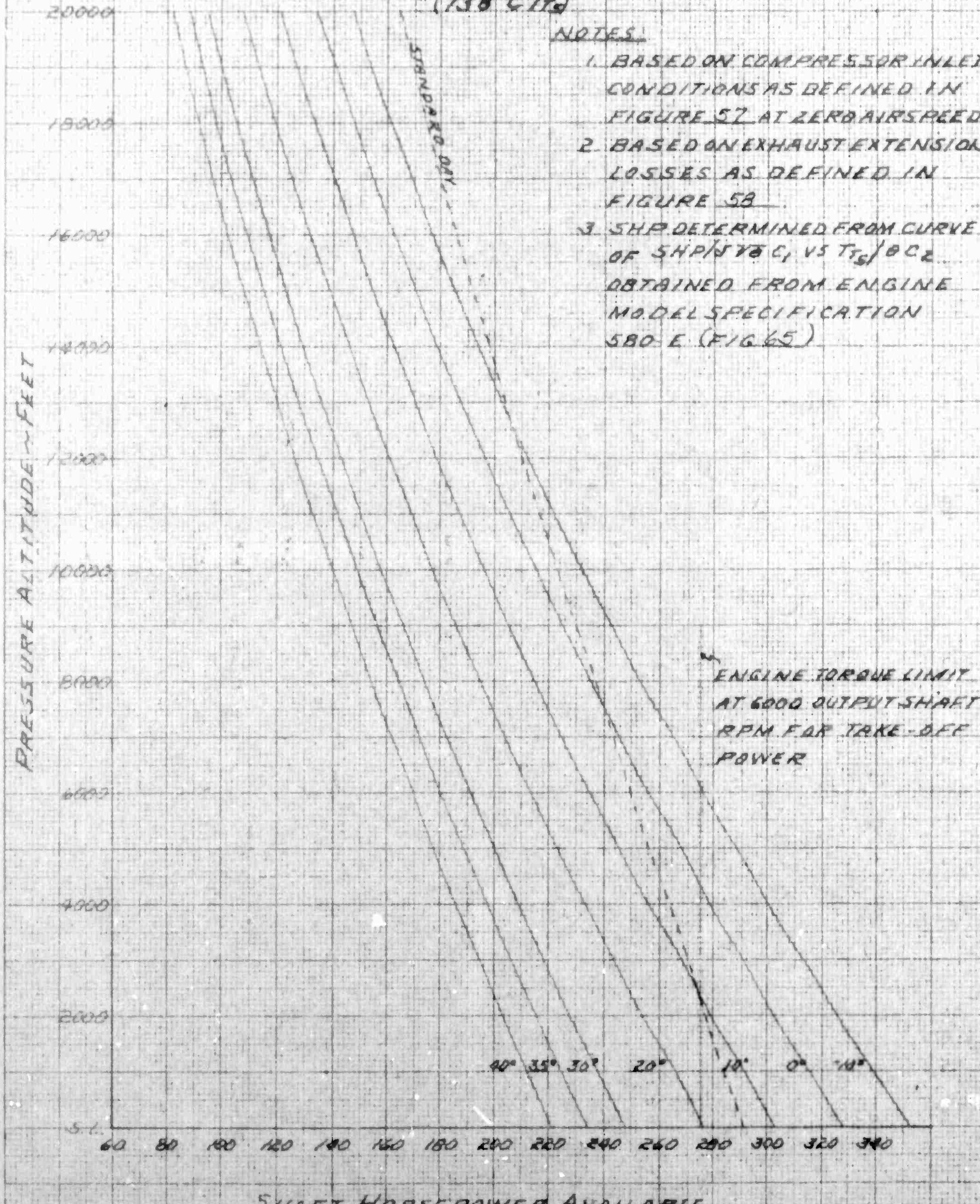


FIGURE NO. 56
SHAFT HORSEPOWER AVAILABLE
OH-4A USA S/N 62-4202
T63-A-5 ENGINE
MAXIMUM CONTINUOUS POWER
(693°C T_{T5})

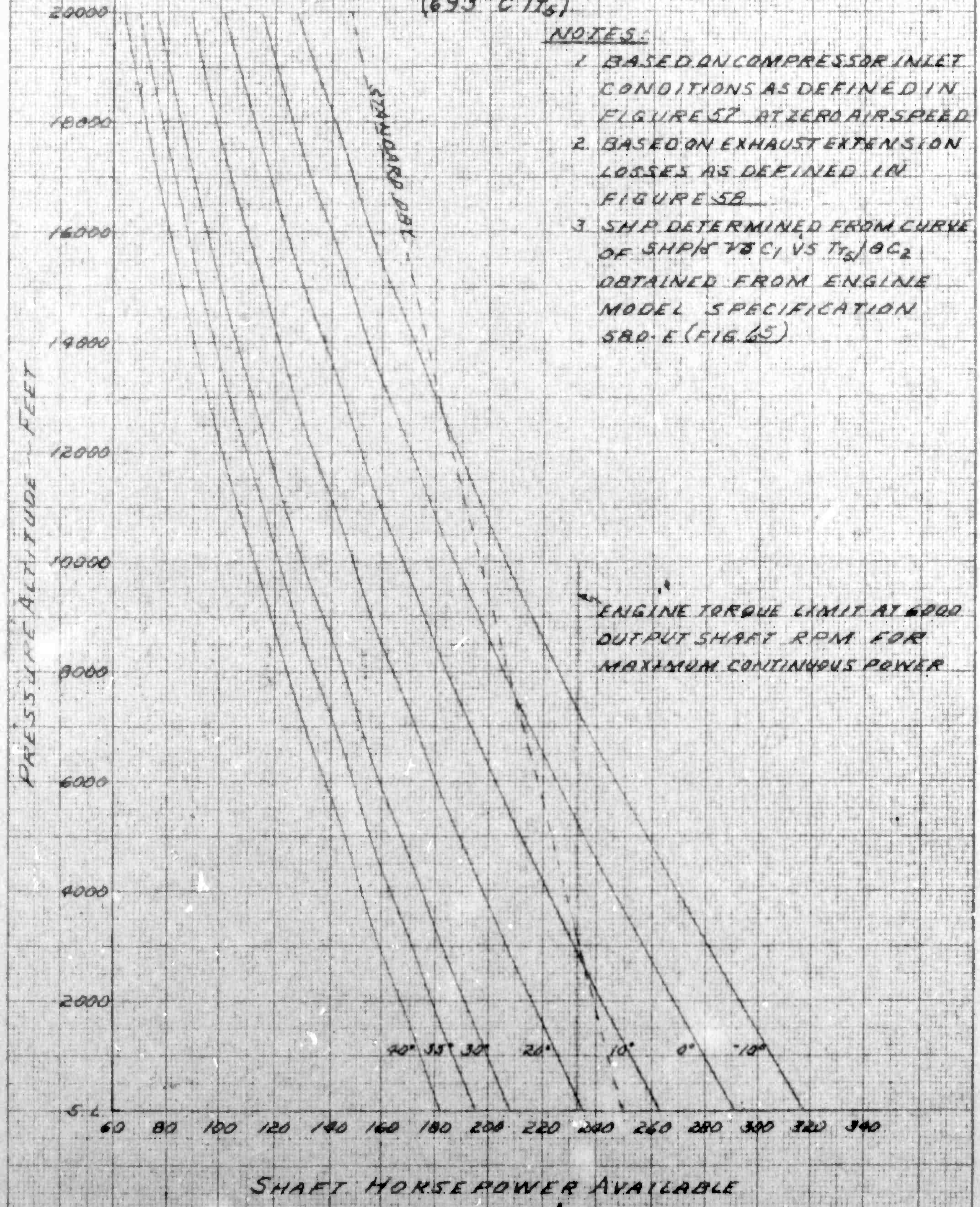


FIGURE No. 57
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

NOTES:
1. P_{T_2} IS COMPRESSOR INLET TOTAL PRESSURE
2. POINTS WERE OBTAINED DURING STABILIZED FLIGHT REF. FIGURES 36 THROUGH 51.

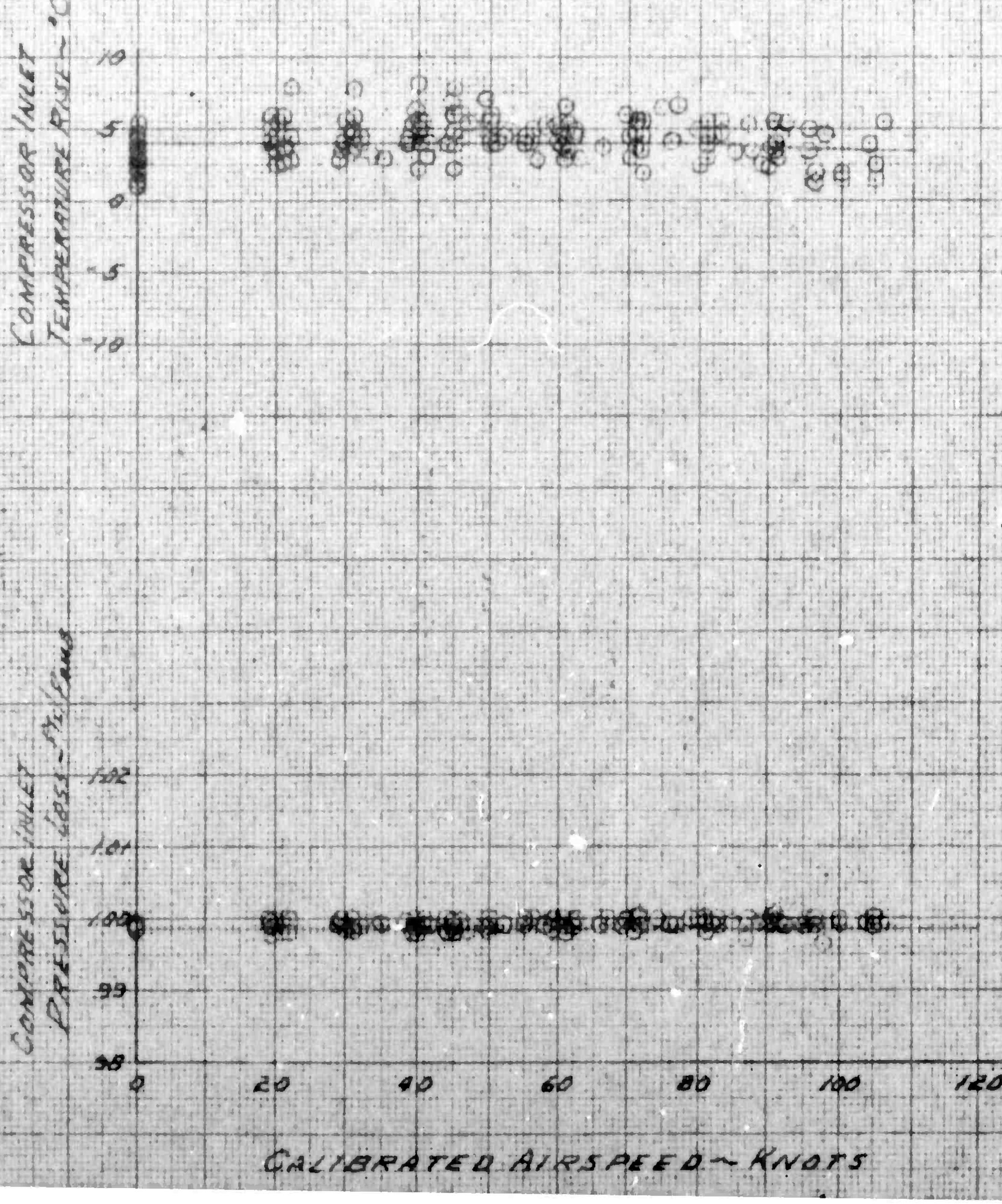


FIGURE No. 58
EFFECT OF BELL EXHAUST EXTENSION ON
T63-A-5 EXHAUST STATIC PRESSURE

OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

NOTES:
1. BASED ON ALLISON TEST STAND DATA.
2. P_{S7} = STATIC PRESSURE AT EXIT OF ENGINE EXHAUST COLLECTOR.
3. P_{AMB} = STATIC PRESSURE AT EXIT OF BELL EXHAUST EXTENSION.
4. θ BASED ON COMPRESSOR INLET TOTAL TEMPERATURE.
5. 100% N_1 = 51,120 RPM.

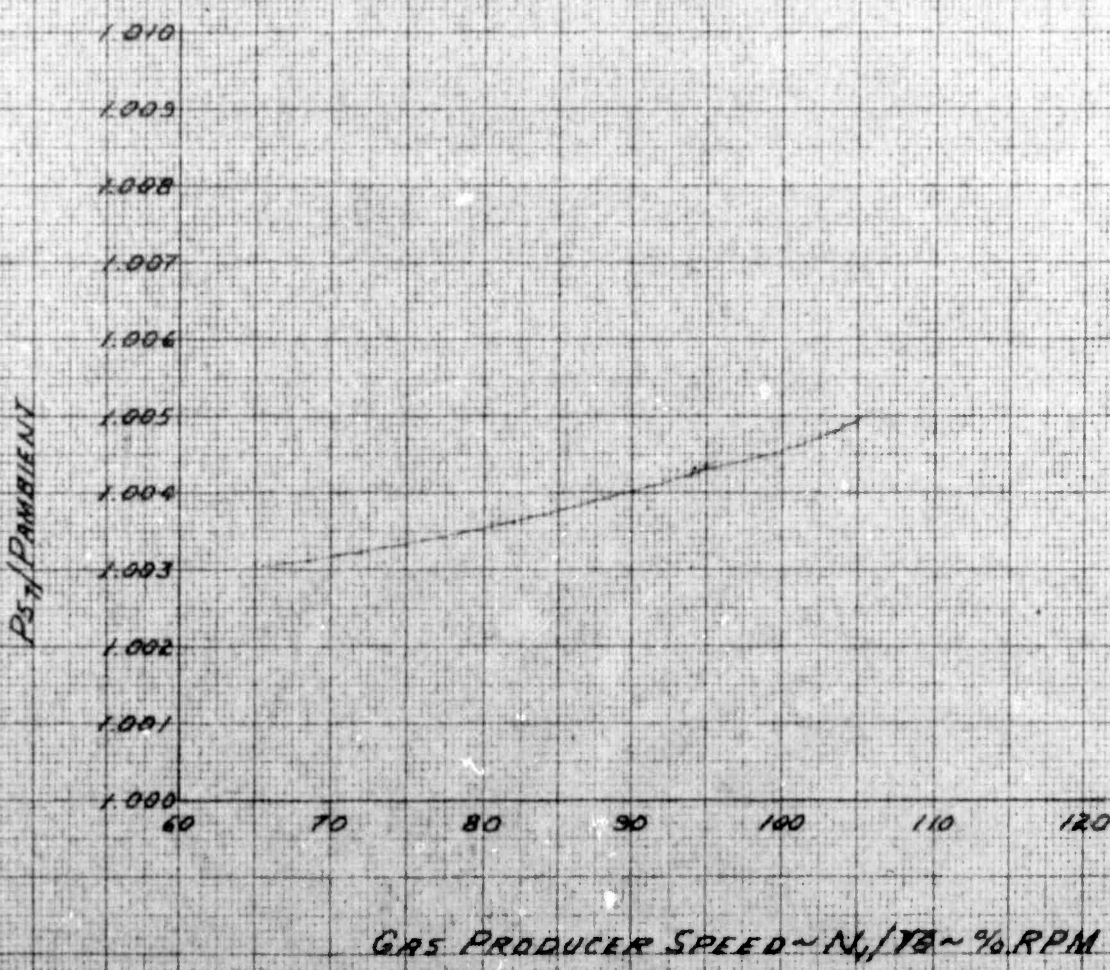
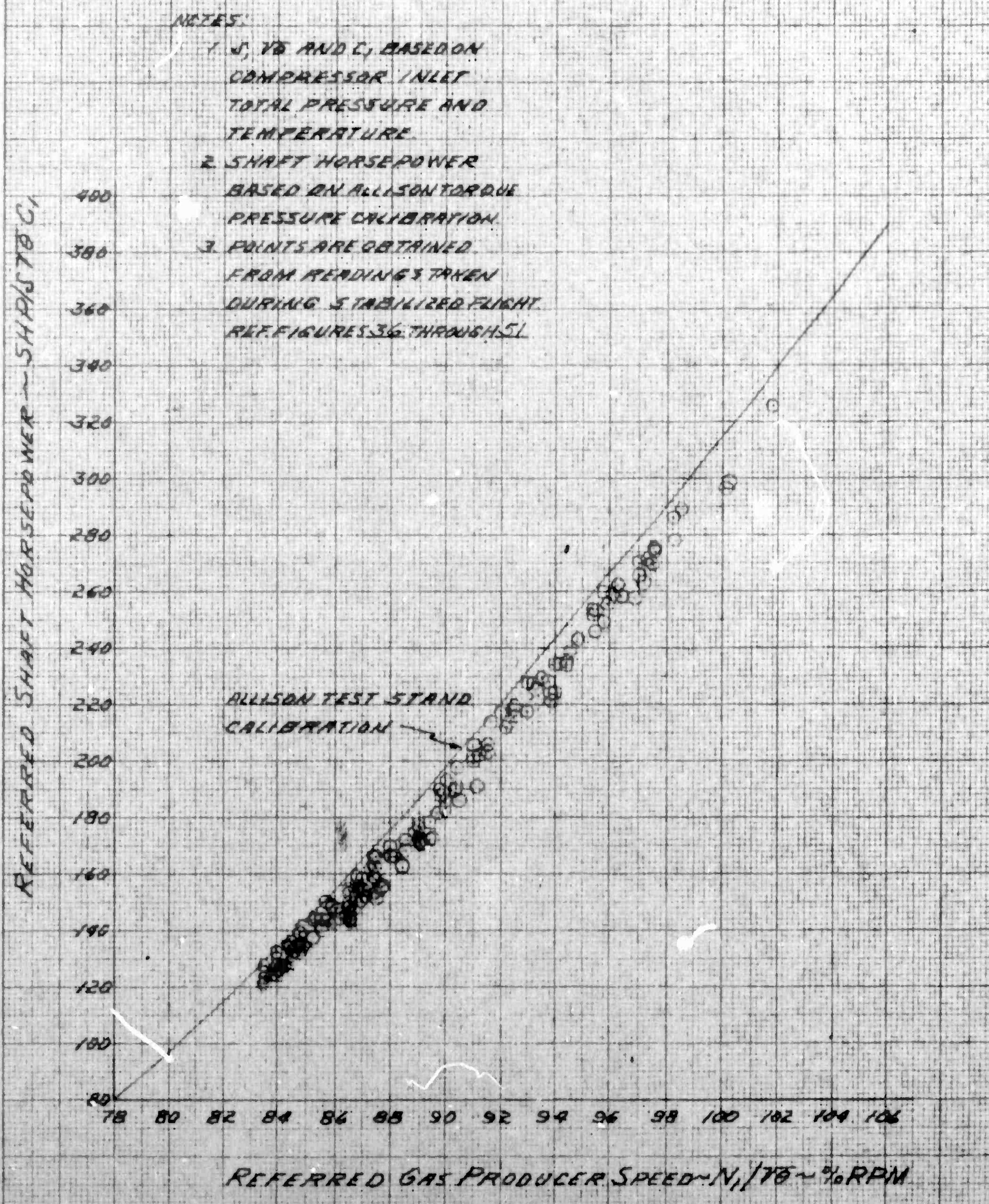


FIGURE NO 59
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040



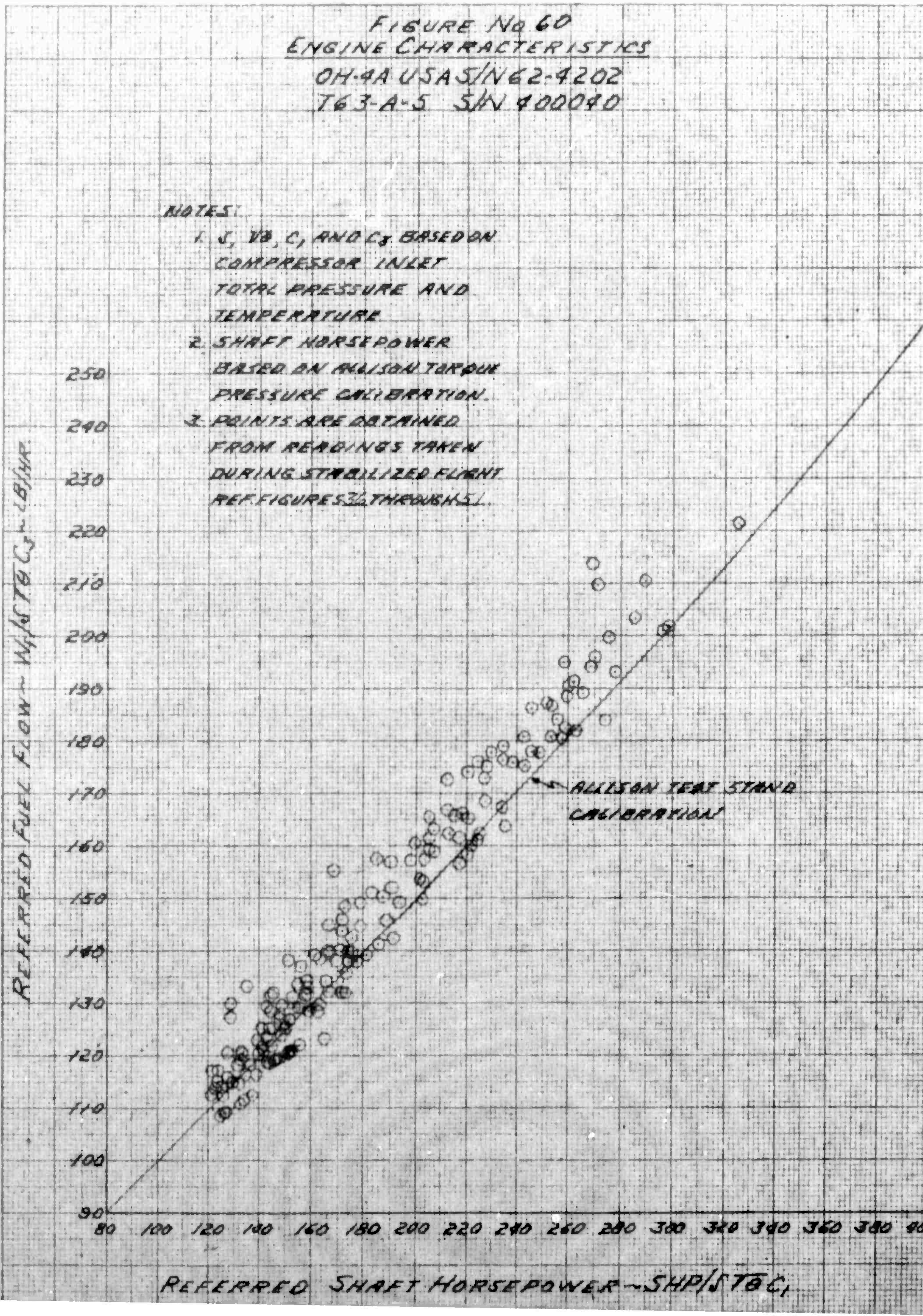


FIGURE No 60
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040
NOTES:
1. δ, √θ, C, AND C₃ BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE
2. SHAFT HORSEPOWER BASED ON ALLISON TORQUE PRESSURE CALIBRATION.
3. POINTS ARE OBTAINED FROM READINGS TAKEN DURING STABILIZED FLIGHT REF FIGURES 36 THROUGH 51.
ALLISON TEST STAND CALIBRATION
REFERRED FUEL FLOW ~ $W_f/\delta\sqrt{\theta}\,C_3$ ~ LB/HR
REFERRED SHAFT HORSEPOWER ~ $SHP/\delta\sqrt{\theta}\,C_1$
90
100
110
120
130
140
150
160
170
180
190
200
210
220
230
240
250
80
100
120
140
160
180
200
220
240
260
280
300
320
340
360
380
400

FIGURE NO. 61
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

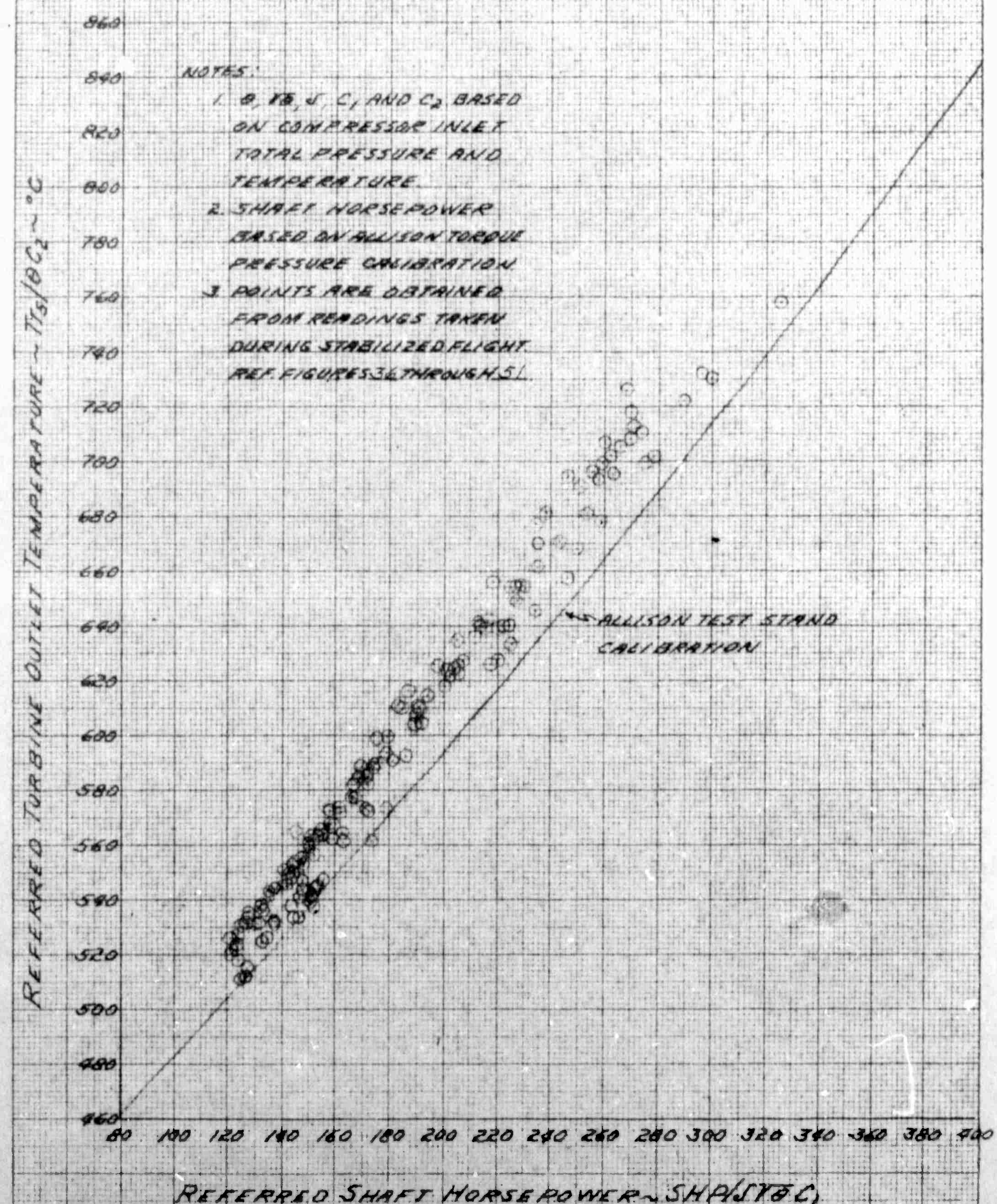
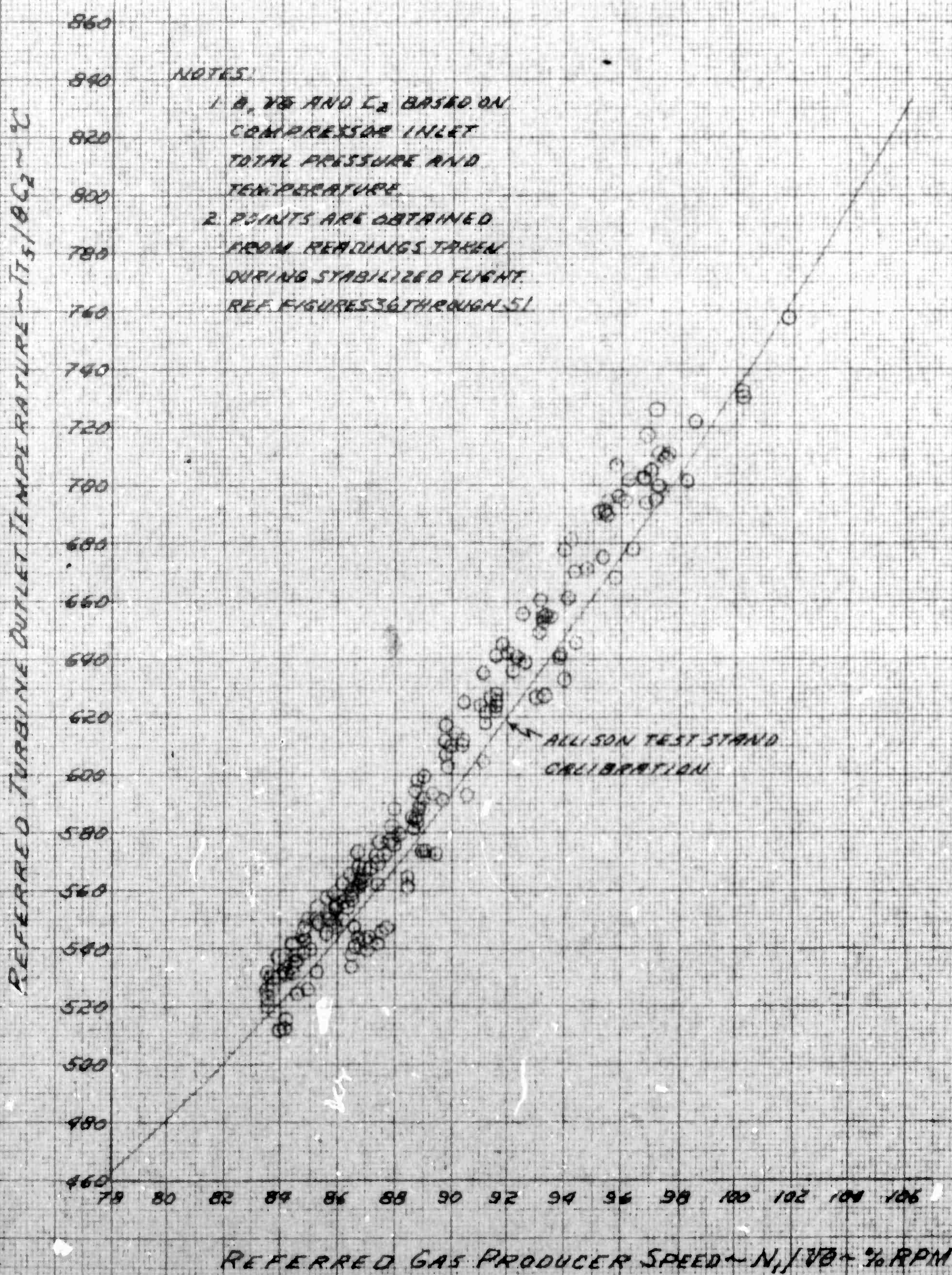
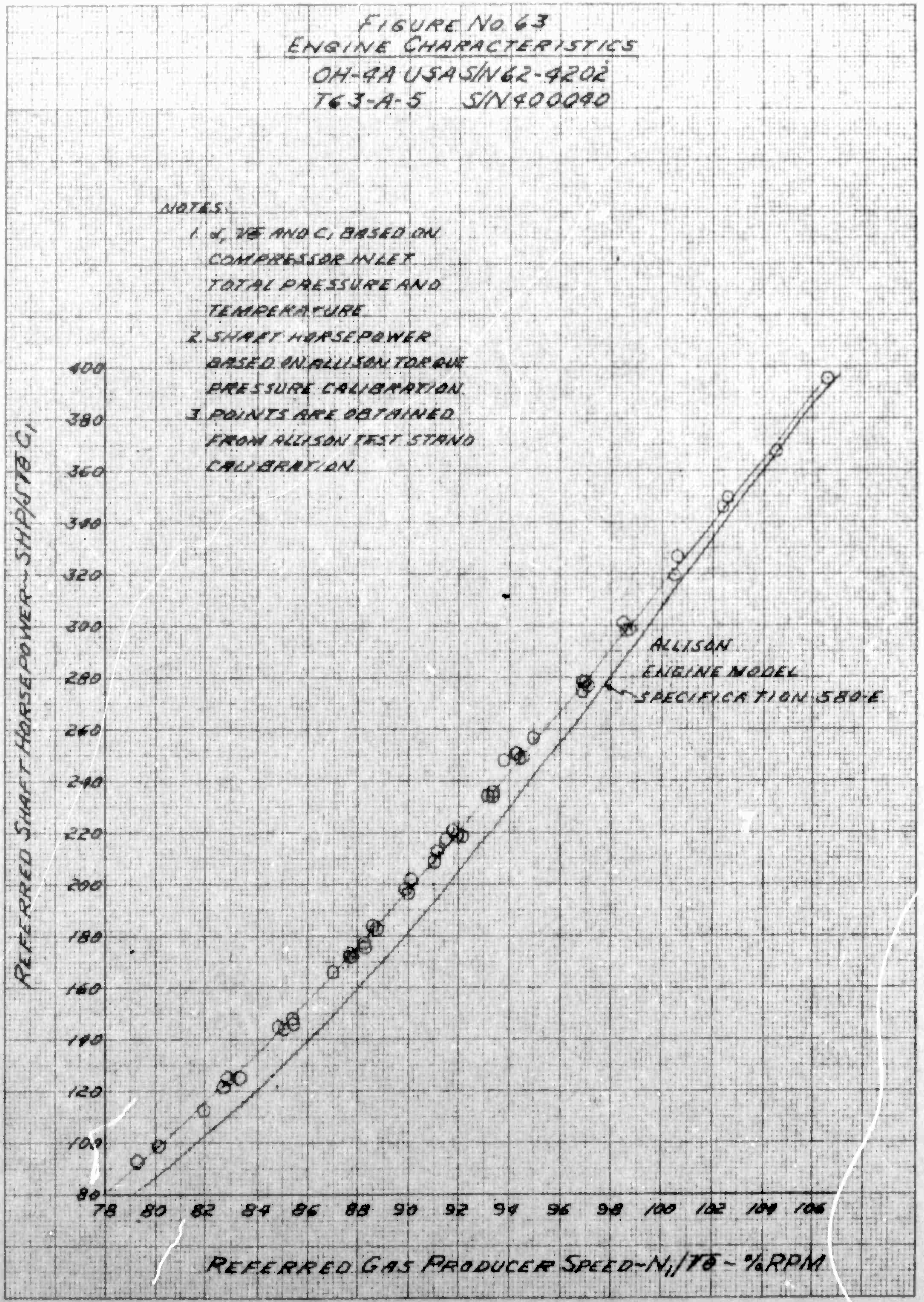


FIGURE NO. 62
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040



FOR OFFICIAL USE ONLY



FOR OFFICIAL USE ONLY

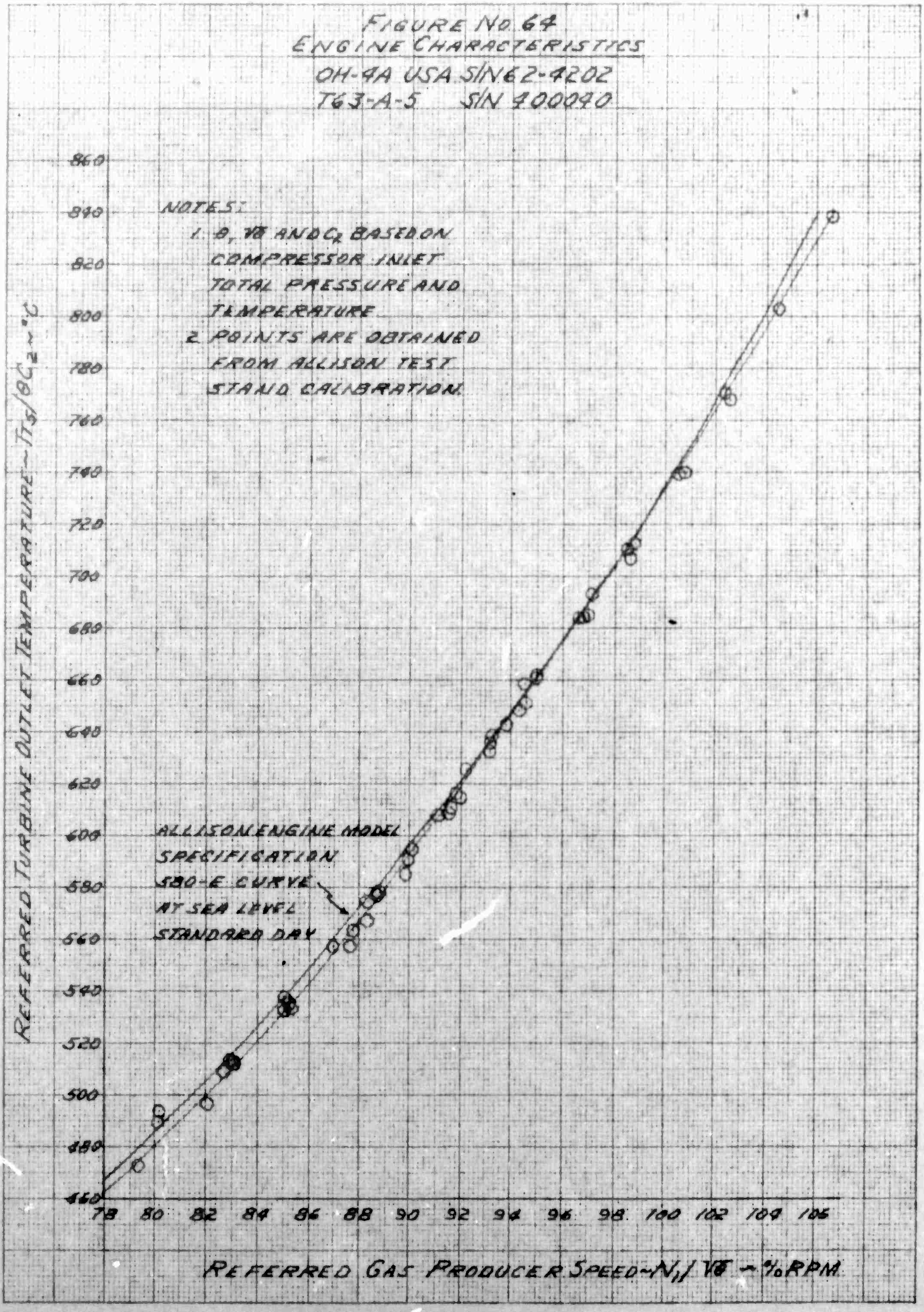
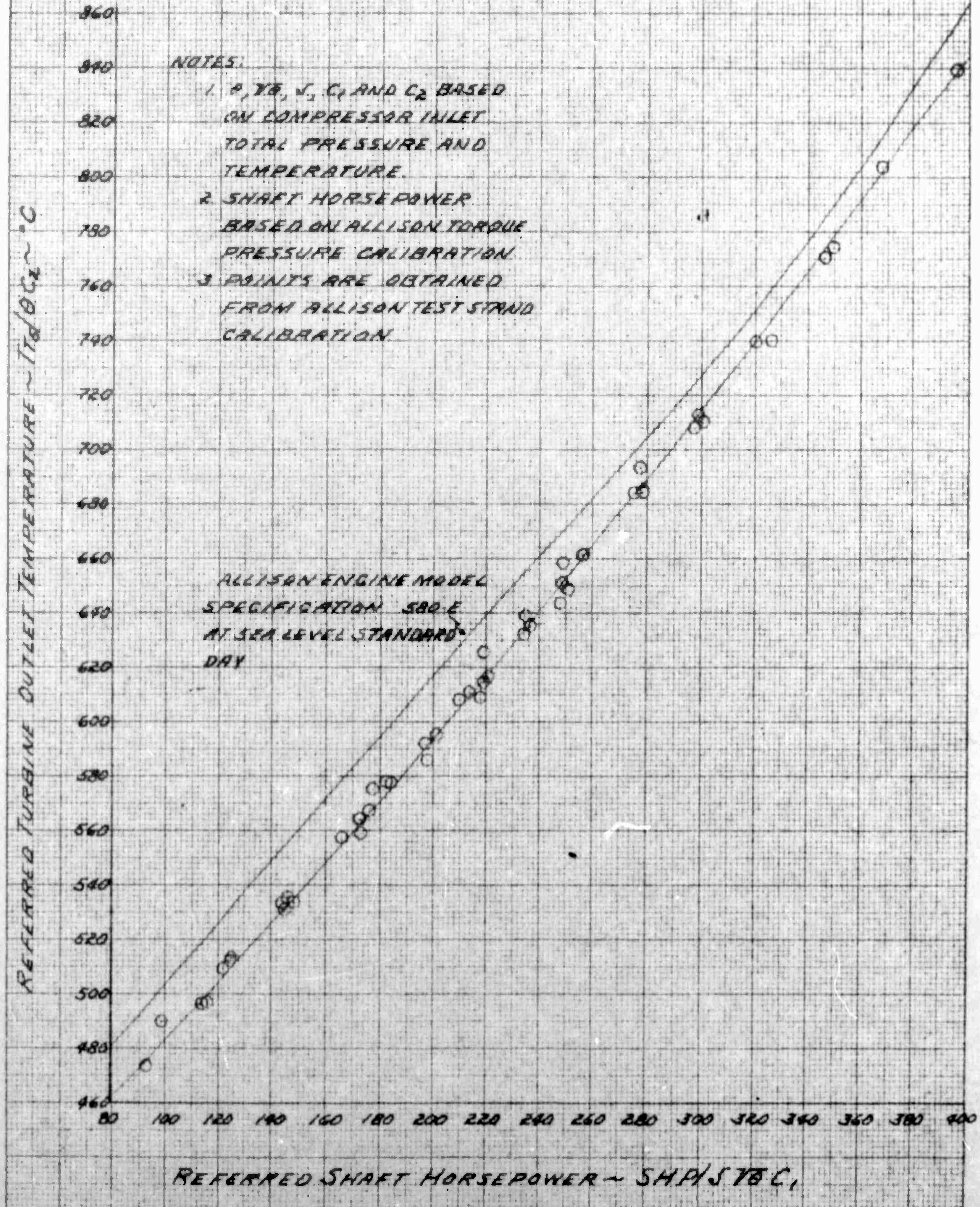
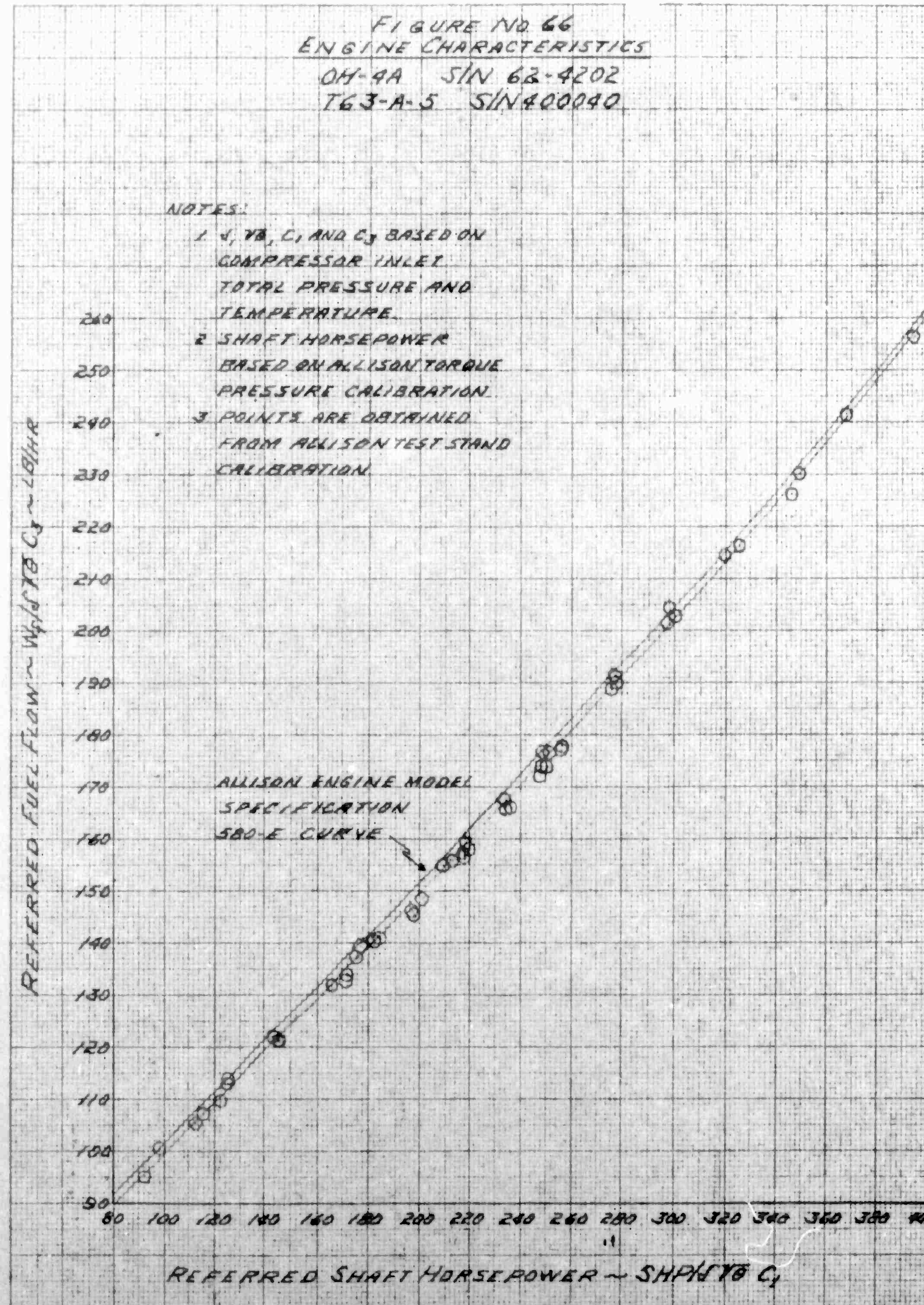


FIGURE NO. 64
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040
NOTES:
1. θ, √θ AND δC_2 BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE
2. POINTS ARE OBTAINED FROM ALLISON TEST STAND CALIBRATION.
ALLISON ENGINE MODEL SPECIFICATION 580-E CURVE AT SEA LEVEL STANDARD DAY
REFERRED TURBINE OUTLET TEMPERATURE ~ $T_{T5}/\theta C_2$ ~ °C
860
840
820
800
780
760
740
720
700
680
660
640
620
600
580
560
540
520
500
480
460
78
80
82
84
86
88
90
92
94
96
98
100
102
104
106
REFERRED GAS PRODUCER SPEED ~ $N_1/\sqrt{\theta}$ ~ % RPM

FIGURE No 65
ENGINE CHARACTERISTICS
OH-4A USA S/N 62-4202
T63-A-5 S/N 400040

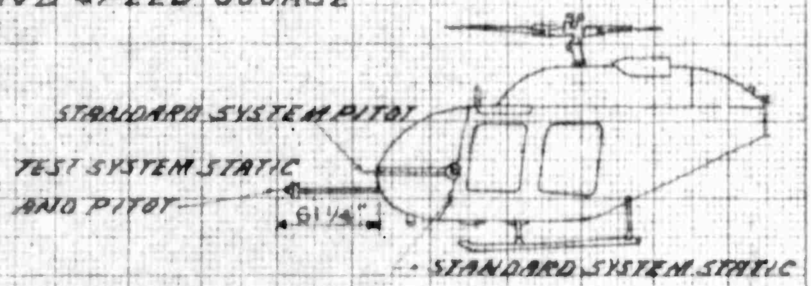


FOR OFFICIAL USE ONLY



FOR OFFICIAL USE ONLY

FIGURE NO. 67
AIRSPEED CALIBRATION
OH-4A USA S/N 62-4202
STANDARD AND BOOM SYSTEM
GROUND SPEED COURSE



SYMBOL	SYSTEM	AVERAGE DENSITY ALTITUDE~FT.	AVERAGE GROSS WEIGHT~LB	AVERAGE C.G.~IN.	ROTOR RPM
○	STANDARD	1260	2550	102.0	394
□	BOOM	1260	2550	102.0	394

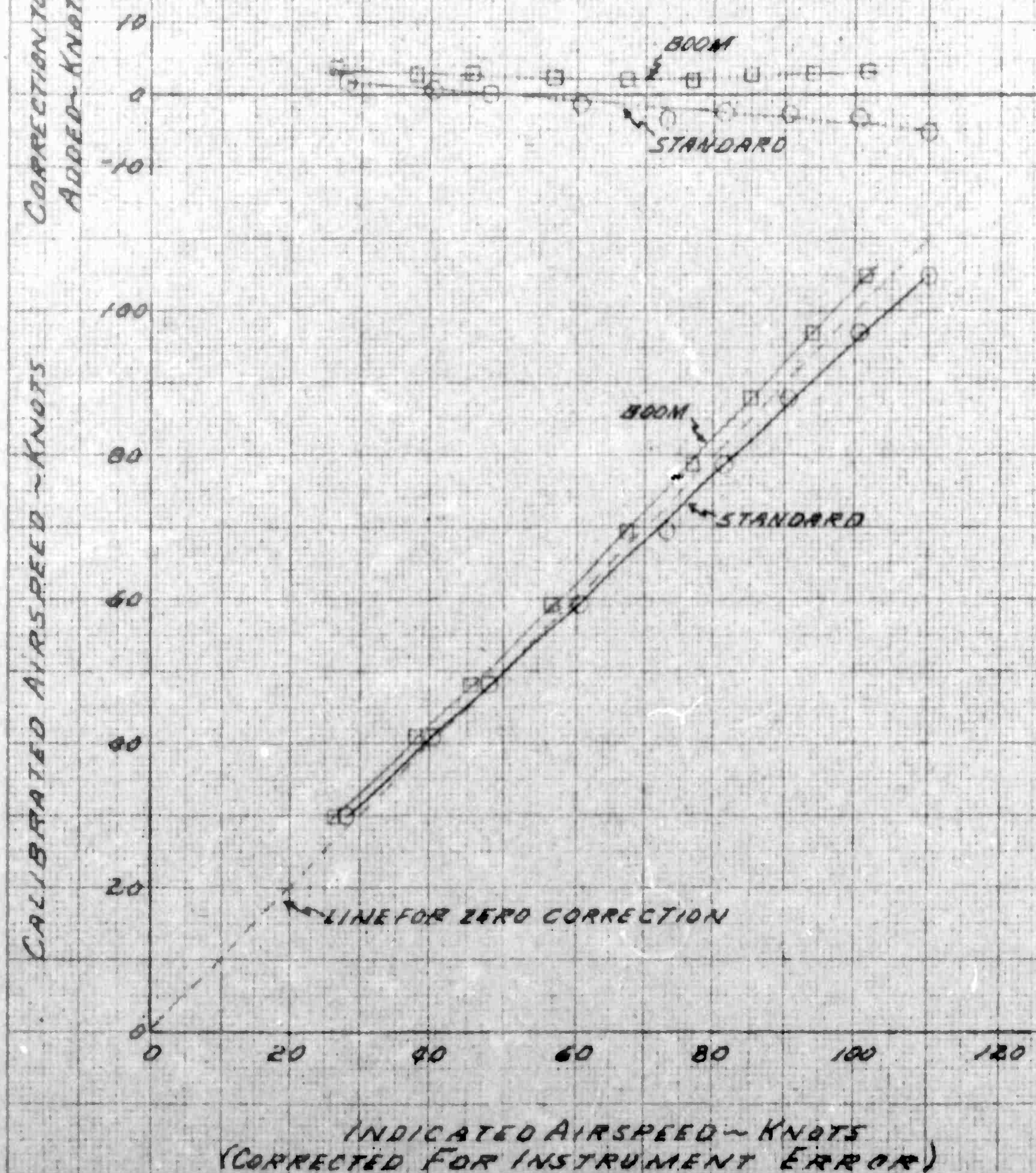
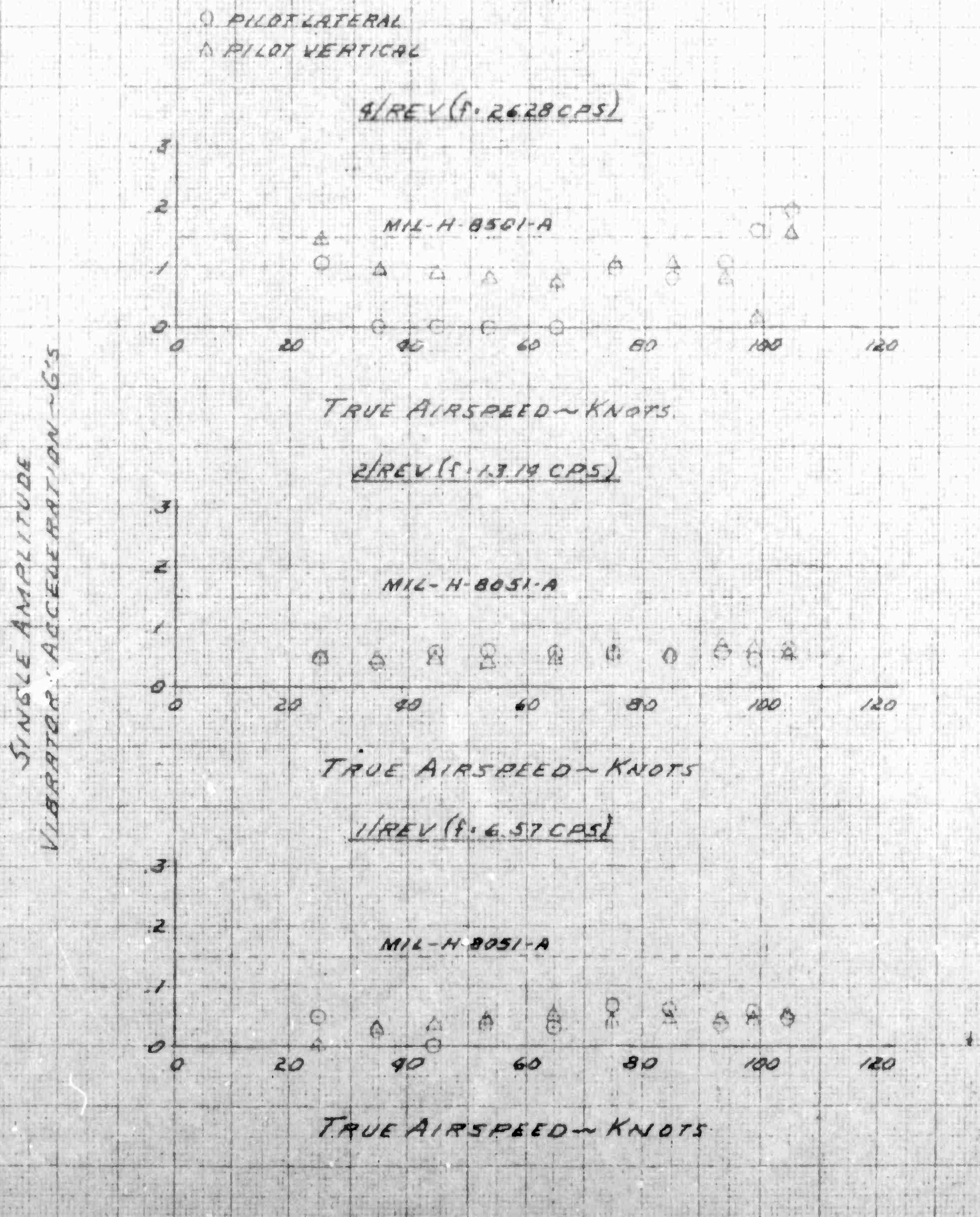


FIGURE NO. 68
VIBRATION CHARACTERISTICS
OH-4A USA S/N 62-4202

ROTOR RPM = 394
GROSS WEIGHT = 2431 LB.
DENSITY ALTITUDE = 827 FT.
C.G. = STATION 102.15 (MID)



APPENDIX II

GENERAL AIRCRAFT INFORMATION

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

1. Sources of Information

See Part I, Section 3, Appendix II.

2. Description of Aircraft and Systems

2.1 Aircraft Design Data

See Part I, Section 3, Appendix II.

2.2 Aircraft Systems

2.2.1 Electrical System

See Part I, Section 3, 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-4A, 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 centigrate (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 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, Section 3, Appendix II.

2.2.4 Fuel System

See Part I, Section 3, Appendix II.

2.2.5 Control System

See Part I, Section 3, Appendix II.

3.0 TIA Limitations

See Part I, Section 3, Appendix II.

4.0 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 1574 pounds with the longitudinal C.G. located at Station 108.4.

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-4A, USA S/N 62-4202, 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 Section 1, paragraph 1.1.o). With these corrections, the empty weight used was 1596 pounds (full oil and trapped fuel). Oil capacity was 9-1/2 pounds.

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

Normal Gross Weight = Empty weight + useful load

Empty Weight = 1596 pounds

Useful Load = Pilot - 200 pounds

Cargo - 400 pounds

Fuel necessary to meet normal gross weight specified in OH-4A Model Specification, paragraph 3.2.1 - 377 pounds

OH-4A Model Specification, paragraph 3.2.1 states that the normal gross weight (performance guarantee weight) shall not exceed 2450 pounds ± 5 percent; therefore, helicopter performance was calculated at 2573 pounds (2450 + 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	-	8.1 pounds
Anti-collision light	-	4.0 pounds
Ground handling wheels	-	32 pounds
XM-7 armament system (full ammunition)	-	375 pounds
XM-8 armament system (full ammunition)	-	325 pounds

After installation of test instrumentation, the helicopter was again weighed. The basic weight (full oil and trapped fuel) was 1867 pounds with the longitudinal C.G. location at Station 105.95.

5.0 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; California Division of Lockheed Aircraft Corporation; Douglas Aircraft Missile 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. 67, 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 (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

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'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 existance of this margin strengthened the T63 manufacturers 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 first 92:15 hours of operation, no detectable deterioration of engine performance occurred in the T63-A-5 engine installed in OH-4A USA S/N 62-4202. At the end of that period, the compressor section of the engine was removed to incorporate an Engineering Order. Because dirt deposits were found, the compressor was cleaned at that time. Following the cleaning, no marked improvement of performance was noted. For nearly all of the period prior to cleaning, the testing was conducted at altitudes greater than 500 feet above the ground, except for normal takeoffs and landings from hard surfaced areas.

The compressor of the engine installed in the OH-4A, USA S/N 62-4204, required cleaning four times during this evaluation so that full power output could be maintained. The compressor was cleaned at period 91:35, 132:25, 140:40 and 145:15 hours of engine operation.

Each time cleaning was required, the preceding 3 to 4 hours of operation were performed under dusty atmospheric conditions. The first cleaning was necessary after approximately 4 hours of operation under conditions in which fine particals of dust were suspended in the atmosphere to an altitude of approximately 10,000 feet, forming a "hazy" condition. The last three cleanings were required following 2 to 3 hours of continuous operation in hovering, sideward and rearward flight, in close proximity to the ground at an unprepared site.

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