UNCLASSIFIED

AD NUMBER
AD875793
LIMITATION CHANGES
TO: **Approved for public release; distribution is unlimited.**
FROM: **Distribution: Further dissemination only as directed by USAAVSCOM Attn: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166, SEP 1970, or higher DoD authority.**
AUTHORITY
USAAVSCOM ltr 12 Nov 1973

THIS PAGE IS UNCLASSIFIED

AD____________

RDTE PROJECT NO.
USAAVSCOM PROJECT NO. 68-30
USAASTA PROJECT NO. 68-30

AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST PRODUCTION OH-58A HELICOPTER UNARMED AND ARMED WITH XM27E1 WEAPON SYSTEM

PERFORMANCE

FINAL REPORT

GEORGE M. YAMAKAWA
PROJECT ENGINEER

JOSEPH C. WATTS
PROJECT OFFICER/PILOT

SEPTEMBER 1970

This document may be further distributed by any holder only with specific prior approval of the CG, USAAVSCOM, ATTN: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166.

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

DISCLAIMER NOTICE

The findings in this report are not to be construed as an official Department of the Army position unless so designated by other authorized documents.

DDC AVAILABILITY NOTICE

~~US military agencies may obtain copies of this report directly from DDC. Other qualified users shall request through the Commanding General, US Army Aviation Systems Command (USAAVSCOM), ATTN: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166.~~

REPRODUCTION LIMITATIONS

Reproduction of this document in whole or in part is prohibited except with permission obtained through the Commanding General, USAAVSCOM, ATTN: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166. DDC is authorized to reproduce the document for United States Government purposes.

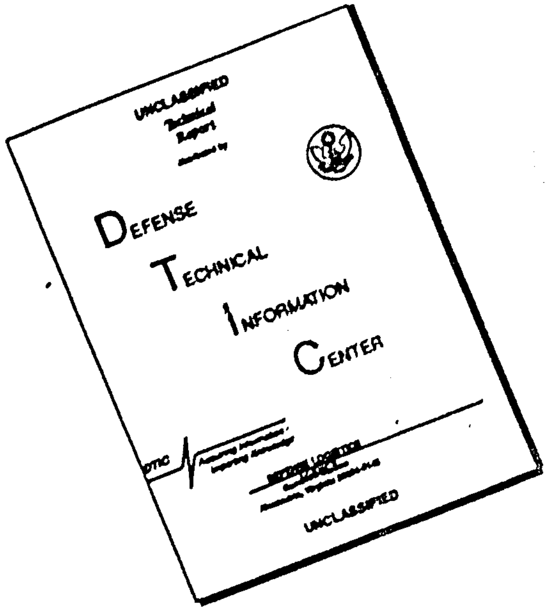
DISPOSITION INSTRUCTIONS

Destroy this report when it is no longer needed. Do not return it to the originator.

TRADE NAMES

The use of trade names in this report does not constitute an official endorsement or approval of the use of the commercial hardware and software.

DISCLAIMER NOTICE



THIS DOCUMENT IS BEST QUALITY AVAILABLE. THE COPY FURNISHED TO DTIC CONTAINED A SIGNIFICANT NUMBER OF PAGES WHICH DO NOT REPRODUCE LEGIBLY.

RDTE PROJECT NO.
USAAVSCOM PROJECT NO. 68-30
USAASTA PROJECT NO. 68-30

AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST

PRODUCTION OH-58A HELICOPTER UNARMED AND ARMED WITH XM27E1 WEAPON SYSTEM

PERFORMANCE

FINAL REPORT

GEORGE M. YAMAKAWA	JOSEPH C. WATTS
PROJECT ENGINEER	PROJECT OFFICER/PILOT

SEPTEMBER 1970

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

ABSTRACT

Performance tests were conducted on a production OH-58A helicopter to determine compliance with performance guarantees outlined in the detail specification and approved revisions, and also to provide information for the operator's manual. Testing was performed by the US Army Aviation Systems Test Activity, Edwards Air Force Base, California, during the period between 19 August 1969 and 13 January 1970. The testing consisted of 99 flights totaling 62.5 productive hours. The OH-58A met or exceeded all of the contractual performance guarantees. The engine inlet loses with the particle separator installed exceeded the limits of the detail specification. A single flight at cold temperatures indicated that level flight performance is significantly affected by compressibility. Although all contractual hover performance guarantees were met, the helicopter could not hover out of ground effect on a 95°F day at gross weights greater than 2930 pounds with the particle separator installed. Under moderate temperature conditions, the performance capabilities of the OH-58A are satisfactory for mission accomplishment. It is recommended that consideration be given to increasing the utility of the OH-58A by installing a more powerful engine.

FOREWORD

Throughout the performance evaluation, technical support was provided under contract by the airframe manufacturer, Bell Helicopter Company, Fort Worth, Texas; and the engine manufacturer, Allison Division of General Motors Corporation, Indianapolis, Indiana. Instrumentation calibration, emergency fire fighting, scientific photography and medical support were provided by the US Air Force Flight Test Center, Edwards Air Force Base, California.

TABLE OF CONTENTS

Page

This page left blank intentionally

INTRODUCTION

BACKGROUND

1. In 1967, the US Army Aviation Systems Test Activity (USAASTA) conducted a preliminary evaluation of a production Model 206A (JetRanger) helicopter manufactured by Bell Helicopter Company. A contract was awarded to Bell Helicopter Company in 1968 to produce for the US Army a Light Observation Helicopter (LOH) designated the OH-58A. Authority for USAASTA to conduct engineering flight tests of a production OH-58A was issued by the Project Manager, US Army Materiel Command in a test directive 7 August 1968 (ref 1, app I). Performance tests were conducted at Edwards Air Force Base, California, and at auxiliary test sites near Bakersfield and Bishop, California.

TEST OBJECTIVES

2. The objectives of the test program were as follows:

a. To determine compliance with detail specification guarantees (ref 2, app I).

b. To provide performance data, with and without the XM27E1 armament system installed, for incorporation into the operator's manual and other publications.

DESCRIPTION

3. The OH-58A Light Observation Helicopter employs a single main rotor and an antitorque tail rotor of the two-bladed, semirigid, teetering type. The tail rotor also has a delta-three coupling. The cockpit provides side-by-side seating for a crew of two (pilot and copilot/observer), and the cargo compartment has provisions for two passengers. Dual flight controls are provided. The cyclic and collective controls are of the hydraulically boosted, irreversible type, and the antitorque control is unboosted. The main landing gear is of the fixed, energy-absorbing skid type. The helicopter is powered by an Allison T63-A-700 free gas turbine engine with a takeoff power rating of 317 shaft horsepower (shp) under sea-level (SL), standard-day, uninstalled conditions. The main transmission has a rating of 270 shp (maximum continuous) with a takeoff power limit of 317 shp (5-minute rating). More detailed aircraft information may be found in appendix II.

4. The XM27E1 armament system consists of one XM134 high-rate 7.62 millimeter (mm) gun (GAU-2B/A) with mount, feed system and ammunition boxes, and one XM70E1 sight for pilot operation. The weapon system is mounted on the left side of the helicopter near the longitudinal center of gravity (cg). The XM134 gun is adjustable in elevation from 5 degrees above to 20 degrees below waterline zero and is operated by either the pilot or copilot/observer.

SCOPE OF TEST

5. The OH-58 was evaluated with respect to its guaranteed mission capabilities as defined in the detail specification. The flight restrictions and operating limitations observed during this evaluation were as specified in the Federal Aviation Administration (FAA) approved operator's manual with exceptions specifically approved by the US Army Aviation Systems Command (USAAVSCOM).

6. Performance tests were conducted in both the armed and unarmed configurations, primarily with all doors installed. In order to determine the effect of door removal on level flight performance, tests were also conducted in the armed configuration with various door combinations. Gross weight was varied from the lightest obtainable to the maximum design weight. A mission cg location at an approximate fuselage station (FS) of 107 inches was used for all tests except for two level-flight speed-power polars that were flown to determine the effect of cg change. Additional tests were conducted to determine the increase in power required for level flight with sideslip, as well as level flight at higher main rotor tip Mach numbers. One flight was conducted with the particle separator removed in order to determine the difference in engine inlet losses. All other tests were flown with the particle separator installed.

7. A total of 99 flights were conducted, consisting of 62.5 productive hours, at Edwards Air Force Base, Bakersfield, and Bishop, California.

METHODS OF TEST

8. Flight test methods used for data acquisition are briefly described in each subtest section of this report and also in the test plan (ref 3, app I). Appendix III outlines the reduction methods used to analyze and evaluate the data in order to determine performance capabilities and compliance with contractual guarantees. All tests were conducted under nonturbulent atmospheric conditions to preclude uncontrolled disturbances influencing the results.

4. The XM27E1 armament system consists of one XM134 high-rate 7.62 millimeter (mm) gun (GAU-2B/A) with mount, feed system and ammunition boxes, and one XM70E1 sight for pilot operation. The weapon system is mounted on the left side of the helicopter near the longitudinal center of gravity (cg). The XM134 gun is adjustable in elevation from 5 degrees above to 20 degrees below waterline zero and is operated by either the pilot or copilot/observer.

SCOPE OF TEST

5. The OH-58 was evaluated with respect to its guaranteed mission capabilities as defined in the detail specification. The flight restrictions and operating limitations observed during this evaluation were as specified in the Federal Aviation Administration (FAA) approved operator's manual with exceptions specifically approved by the US Army Aviation Systems Command (USAAVSCOM).

6. Performance tests were conducted in both the armed and unarmed configurations, primarily with all doors installed. In order to determine the effect of door removal on level flight performance, tests were also conducted in the armed configuration with various door combinations. Gross weight was varied from the lightest obtainable to the maximum design weight. A mission cg location at an approximate fuselage station (FS) of 107 inches was used for all tests except for two level-flight speed-power polars that were flown to determine the effect of cg change. Additional tests were conducted to determine the increase in power required for level flight with sideslip, as well as level flight at higher main rotor tip Mach numbers. One flight was conducted with the particle separator removed in order to determine the difference in engine inlet losses. All other tests were flown with the particle separator installed.

7. A total of 99 flights were conducted, consisting of 62.5 productive hours, at Edwards Air Force Base, Bakersfield, and Bishop, California.

METHODS OF TEST

8. Flight test methods used for data acquisition are briefly described in each subtest section of this report and also in the test plan (ref 3, app I). Appendix III outlines the reduction methods used to analyze and evaluate the data in order to determine performance capabilities and compliance with contractual guarantees. All tests were conducted under nonturbulent atmospheric conditions to preclude uncontrolled disturbances influencing the results.

9. An OH-58A helicopter (S/N 68-16693) was equipped with sensitive, calibrated instrumentation. A detailed list of the test instrumentation is presented in appendix IV. Photographs of the cockpit and cabin instrumentation are found in appendix V. Qualitative pilot comments were used to aid in the analysis of data and to assist in the overall assessment of the performance characteristics of the OH-58A.

CHRONOLOGY

10. The chronology of the OH-58A test program is as follows:

Test directive issued	7 August	1968
Test plan submitted (revised)	May	1969
Test plan approved	July	1969
Test aircraft received	3 July	1969
Tests started	18 August	1969
Calibrated engine installed	23 August	1969
Tests completed	13 January	1970
Draft report submitted	May	1970

RESULTS AND DISCUSSION

GENERAL

11. Test results indicate that the performance capabilities of the OH-58A helicopter are satisfactory for mission accomplishment under moderate temperature conditions. At cold temperatures and high advancing tip Mach numbers, compressibility effects significantly decrease the level flight performance. On a hot day (95°F), the OH-58A helicopter lacks the power available for out of ground effect (OGE) hover capability at gross weights greater than 2930 pounds. All detail specification performance guarantees were met or exceeded. Guarantee compliance was based on fuel-flow and power-available data presented in the T63-A-700 engine model specification (ref 4, app I) without the particle separator installed. This particle-separator-removed condition was a contractual agreement defined in the detail specification. Definitions of configuration I and II gross weights, which were used for all of the guarantee compliances, are summarized in table 1. Table 2 summarizes performance guarantee results at 354 rotor rpm based on the engine inlet losses without the particle separator installed.

12. The test data that are presented in appendix VI represent conditions with the particle separator installed, except for five plots which were used for guarantee compliance and are indicated by the notation "particle separator removed."

Table 1. Guarantee Gross Weights.

Item	Weight (lb)	Configuration I[1]	Configuration II[2]
Guaranteed empty weight	1586	1586	1586
Crew of two	400	400	400
Engine oil	10	10	10
Removable armor	112	112	112
Armor chest protectors	30	30	30
Mission equipment	170	170	170
XM27E1 armament system	106	0	106
2000 rounds of ammunition	128	0	128
Fuel	--	[3]443.9	[4]417.8
Total gross weight		2752	2960

[1]Observation mission.
[2]Scout mission.
[3]Fuel required for a 260 nautical air mile range, sea level, standard day, particle separator removed; 10 percent of initial fuel for reserve, takeoff fuel allowance consisting of 2-minute fuel at normal rated power (conditions as defined in ref 2, app I).
[4]Same conditions as footnote 3 above except for a 230 nautical air mile range.

Table 2. Performance Guarantee Summary
(Particle Separator Removed).

Configuration I[1]		
Item	Guaranteed	Test Results
FAA certified V_{NE}[2], sea level, standard day	120 KCAS[3]	123 KCAS
Power required, 110 KTAS[4], sea level, standard day	255 shp	246 shp
Maximum rate of climb at MRP[5], sea level, standard day	1500 fpm[6]	1815 fpm
Hover ceiling at MRP, OGE, 95°F day	2000 ft	2240 ft
Hover ceiling at MRP, IGE[7], 4-foot skid height, 95°F day	5000 ft	5030 ft
Range at sea level, standard day	260 NAMT[8]	260 NAMT
Endurance at sea level, standard day	3.0 hrs	3.63 hrs
Configuration II[9]		
Range at sea level, standard day	230 NAMT	230 NAMT
Hover ceiling at MRP, OGE, standard day	6000 ft	6270 ft[10]

[1]Observation mission gross weight calculated to be 2752 pounds.
[2]Never exceed airspeed.
[3]Knots calibrated airspeed.
[4]Knots true airspeed.
[5]Military rated power.
[6]Feet per minute.
[7]In ground effect.
[8]Nautical air miles traveled.
[9]Scout mission gross weight, XM27E1 gun system installed, calculated to be 2960 pounds.
[10]Extrapolated from test data.

PERFORMANCE CHARACTERISTICS

Takeoff Performance

13. Takeoff performance tests were conducted to determine the total distance required to clear a 50-foot obstacle under conditions where the OH-58A cannot accomplish a vertical takeoff. Test results are presented in figures 2 through 5, appendix VI, and are summarized in figure 1. All takeoffs were initiated from a stabilized 2-foot hover skid height. Because of the limited time available for this evaluation, only the level acceleration takeoff method was used. This was qualitatively determined to be the optimum method for all operating conditions. Under conditions of large power margins (excess power available over power required), the climb acceleration takeoff method requires a high degree of pilot skill and proficiency inconsistent with normal operational requirements. During the takeoff tests, regardless of method, increased pilot attention was required in order to maintain the desired power setting without exceeding the maximum takeoff power (TOT) limit (749°C).

Hover Performance

14. The hover performance capabilities of the OH-58A were evaluated IGE and OGE at various skid heights and gross weights, both in the clean (unarmed) and armed configurations. All hover performance data were collected using the free-flight method. A premeasured, weighted cord was attached to the skid to precisely determine height above the ground. The hover performance capabilities are summarized and presented in figures 6 and 7, appendix VI, as plots of hover capabilities based on gross weight, skid height and pressure altitude for various temperatures at takeoff and normal rated power. Nondimensional plots of power coefficient (C_P) versus thrust coefficient (C_T) are presented in figures 8 through 14 for skid heights of 2, 4, 10 and 20 feet (IGE) and 50 feet (OGE). These tests were conducted at density altitudes ranging from 235 to 10,110 feet and at gross weights from 2085 to 3055 pounds.

15. All OH-58A hover ceiling performance guarantees were exceeded. As stipulated, the model engine specification power available (without the particle separator installed) was used to determine guarantee compliance. At configuration I gross weight (2752 pounds), hot-day (95°F), OGE conditions, test data indicated a hover ceiling of 2240 feet as compared to a guarantee of 2000 feet. At the same gross weight, IGE (4-foot skid height), results indicated a hot-day (95°F) hover ceiling capability of 5030 feet as compared to a 5000-foot guarantee. In order to determine guarantee compliance for configuration II gross weight (2960 pounds) at standard-day, OGE conditions, the extrapolated portion of the faired curve was used, (fig. 14, app VI). The results indicated a hover ceiling capability of

6270 feet as compared to a guarantee of 6000 feet. Figure A summarizes hover ceiling test data as compared to the performance guarantee.

16. The hover performance of the OH-58A with the particle separator installed was adequate for mission accomplishment under moderate temperature conditions. Standard-day hover performance data indicate a sea-level capability of hovering OGE at gross weights greater than the maximum design gross weight (3000 pounds) at 354 rpm. However, on a hot day (95°F) with the particle separator installed, the OH-58A helicopter will not hover OGE at gross weights greater than 2930 pounds. Installation of the particle separator decreased the T63-A-700 engine power available by 7.5 shp at sea-level, standard-day conditions. A comparison between the hover performance with the particle separator installed and the hover performance with the particle separator removed is illustrated in figure A.

17. Since it is not a requirement of the OH-58A detail specification, the lack of a 95°F, hot-day, OGE hover capability at the configuration II gross weight could not be termed a deficiency or shortcoming of the aircraft. However, experienced test pilots who have served in Southeast Asia affirm the need for a helicopter which meets these design criteria. In combat situations, there is a requirement to maximize an aircraft's usefulness in terms of range, endurance and load-carrying capability. The OH-58A will, therefore, probably be flown in that environment at or near its maximum design gross weight of 3000 pounds and in ambient temperatures often reaching 95°F. The test results (takeoff power, model specification engine, particle separator installed, fig. A) show that at the configuration II gross weight, the OH-58A can hover IGE at pressure altitudes up to 2450 feet on a 95°F day (5100-foot density altitude). This condition will allow mission accomplishment but does not provide the safety or design margin afforded by an OGE hover capability. For a normal, safe operation, a reduced fuel load with a consequent reduction in range and endurance would be required. Considering the possibility of dirty, worn engines, inaccurate calculation of gross weight, and other factors common to an operational environment, it is recommended that the OH-58A not be flown at gross weights in excess of 2930 pounds at density altitudes higher than 2200 feet (sea level, 95°F) except in emergency tactical situations. Gross weight limitations for a range of temperature and altitude conditions should be specified in the operator's manual. Because of the temperature-limited design characteristic of the OH-58A, weight limitations are more critical at high temperature conditions than at high pressure altitudes. For increased utility, it is also recommended that consideration be given to installing a more powerful engine in the OH-58A.

FIGURE A

SUMMARY HOVER PERFORMANCE COMPARISON FOR GUARANTEE COMPLIANCE

OH-58A USA S/N 68-16693

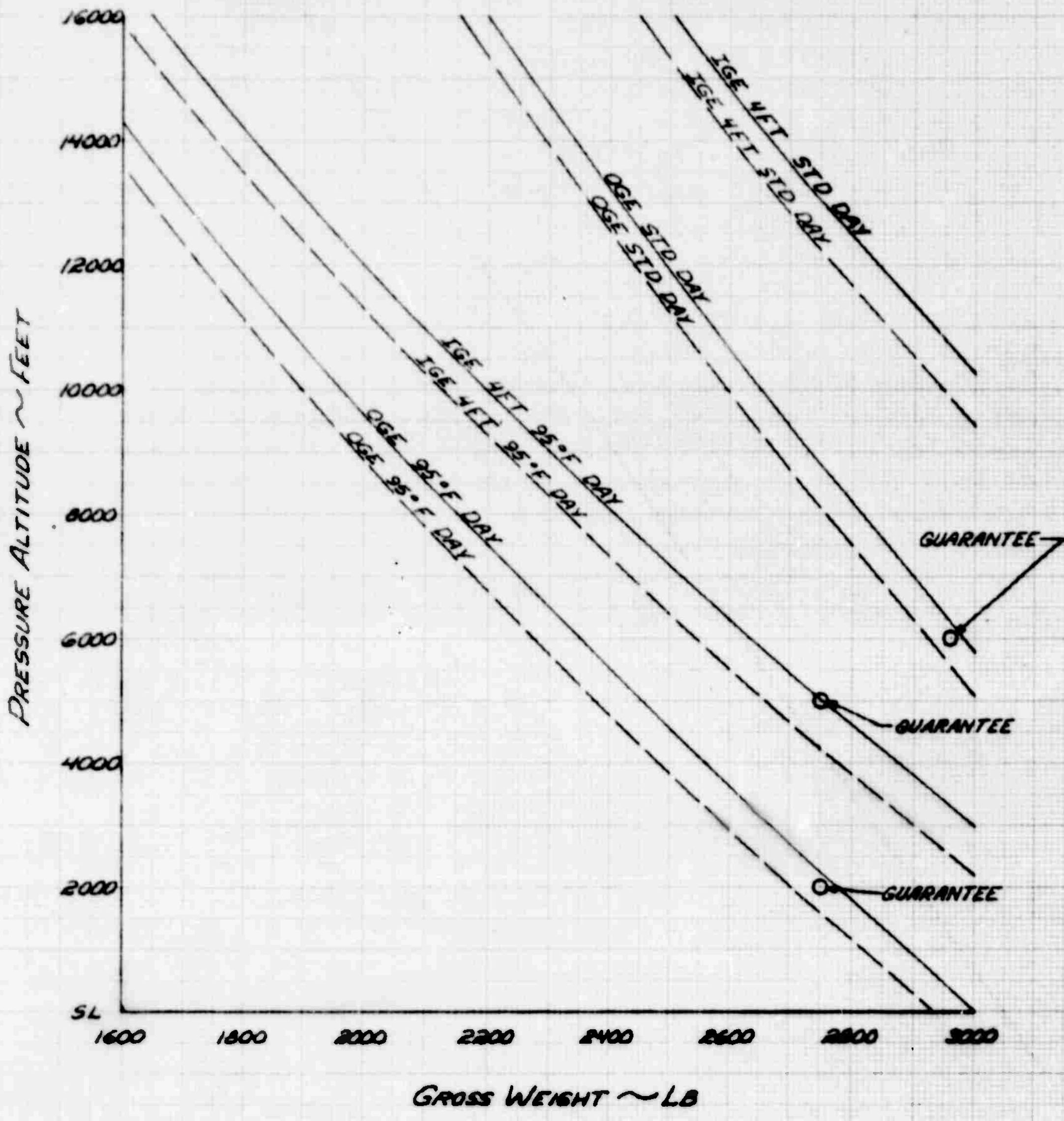
T63-A-700 ENGINE

FREE FLIGHT METHOD

WINDS LESS THAN 3 KNOTS

NOTE:

1. POWER AVAILABLE SPECIFICATION ENGINE OBTAINED FROM FIG. 69 APPENDIX VI.
2. HOVER CAPABILITY OBTAINED FROM FIGS. 11 AND 14 APPENDIX VI.
3. SOLID LINES DENOTE WITHOUT PARTICLE SEPARATOR INSTALLED.
4. DASHED LINES DENOTE WITH PARTICLE SEPARATOR INSTALLED.



Climb Performance

18. Continuous climbs were conducted from sea level to service ceiling to determine the OH-58A climb capabilities and compliance with the guarantees. All climbs were flown at the airspeed schedule for best rate of climb at 354 rotor rpm, and at both takeoff and maximum continuous power settings. The climb schedules were determined from level-flight data at the minimum power required for level flight on a standard day. Although the climb schedule was based on standard conditions, tests were flown with reference to pressure altitude disregarding existing ambient temperature. The data were corrected from test-day conditions to standard-day conditions and also to model specification engine power and fuel flow with the particle separator installed. No attempt was made to correct rate of climb for compressibility since the extent of this effect was not determined. Test results are presented in figures 15 through 20, appendix VI.

19. In order to make a valid comparison, flight test data were corrected to the detail specification guarantee condition. The calculated value shows that the maximum rate of climb at sea level, using takeoff power (without the particle separator installed), was approximately 1815 fpm. This exceeded the guaranteed rate of climb of 1500 fpm by 21 percent at the configuration I gross weight (2752 pounds). The computation of the compliance guarantee is summarized as follows:

Item	Guarantee Condition[1]	Flight Test Condition[2]
Altitude	SL, standard day	SL, standard day
Gross weight (GW)	2752 lb	2747 lb
Particle separator	Removed	Installed
Power setting	Takeoff	Takeoff
Rotor speed	354 rpm	354 rpm

[1]Conditions defined in detail specification.
[2]Data obtained from figure 15, appendix VI.

Item	Test Value	Source
Rate of climb (R/C)	1750 fpm	Fig. 15, app VI
Shaft horsepower test (SHP_t)	300 shp	Fig. 15, app VI
Shaft horsepower standard (SHP_s)	307.5 shp	Fig. 69, app VI
Power correction factor (K_p)	0.7946	Fig. 21, app VI
Weight correction factor (K_w)	0.94 at 2752 lb	Fig. 22, app VI

Formulas:

$$R/C_s = R/C + \Delta R/C_p + \Delta R/C_w$$

$$\Delta R/C_p = K_p \times \frac{(SHP_s - SHP_t) \times 33{,}000}{GW_t}$$

$$\Delta R/C_w = K_w \times SHP_s \times 33{,}000 \left(\frac{1}{GW_s} - \frac{1}{GW_t}\right)$$

Computation:

$$R/C_s = 1750 + 0.7946 \frac{(307.5 - 300) \times 33{,}000}{2747}$$

$$+ 0.94 \times 307.5 \times 33{,}000 \times \left(\frac{1}{2752} - \frac{1}{2747}\right)$$

$$R/C_s = 1815 \text{ fpm}$$

20. The climb performance of the OH-58A with the particle separator installed was satisfactory for mission accomplishment. The maximum rates of climb (corrected for test weight variation from configuration I gross weight) for takeoff and maximum continuous power at sea level were 1744 and 1350 fpm, respectively. Under the same conditions except for changing the gross weight to that of configuration II (2960 pounds), the maximum rates of climb were 1570 and 1210 fpm, respectively. Table 3 summarizes service ceiling results at 354 rpm, with the particle separator installed, and at standard-day conditions.

Table 3. Service Ceiling Summary.

Takeoff Gross Weight (lb)	Power Setting	Service Ceiling (ft)
2,252	Takeoff	26,170
2,248	Maximum continuous	25,000
2,747	Takeoff	21,800
2,755	Maximum continuous	19,500
2,960	Takeoff	18,380
2,965	Maximum continuous	17,200

21. In addition to the continuous climb tests, two series of sawtooth climbs were conducted to determine correction factors for variation in power (K_p) and gross weight (K_w). These factors were used to correct continuous climb data from test to standard conditions. The first series of sawtooth climbs was flown at constant gross weight and varying power. The second series was flown at a constant power and varying gross weight. The test results are presented in figures 21 and 22, appendix VI. The plots indicate that K_p is equal to 0.7946 for all conditions tested, and K_w varies as a function of gross weight.

Level Flight Performance

22. Tests were conducted to determine airspeed, fuel flow, and power-required relationships to define the level flight performance for combinations of external configuration, gross weight, altitude, and rotor rpm. All flights were conducted at zero sideslip except for one which was flown to determine sideslip effects. Each speed power was conducted at a constant value of gross weight divided by density (W/ρ). This procedure necessitated an increase in altitude for successive data points as fuel was consumed. Tests were conducted

at gross weights ranging from 2180 to 3007 pounds and at density altitudes from 1150 to 15,160 feet. Nondimensional summary plots are presented in figures 23 through 25, appendix VI. Specific range summaries for sea-level, 5000- and 10,000-foot standard-day conditions (to include the effects of configuration change, cg change and sideslip), are summarized in figures 26 through 31. Individual test results are presented graphically in figures 32 through 53.

23. All level-flight performance guarantees were met or exceeded. Figure B compares guarantees to test results for V_{NE} and for power required at 110 KTAS in level flight. As determined by flight test, the power required to maintain 110 KTAS was 9 shp less than the guarantee. Using the maximum available shaft horsepower of a model specification engine (sea level, particle separator removed and measured installation losses included), a V_{NE} which was 3 knots higher than the guarantee was obtained from figure B.

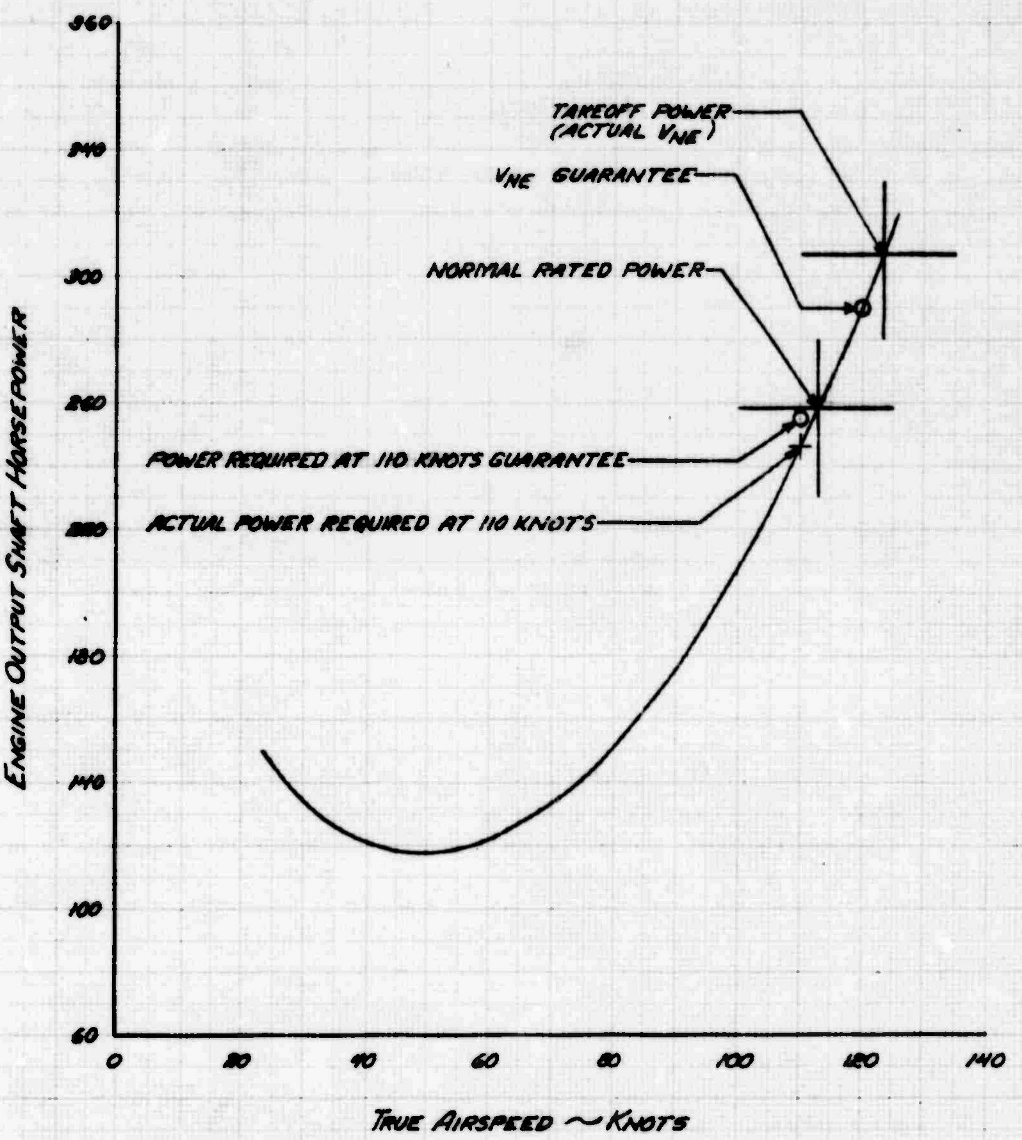
FIGURE B

LEVEL FLIGHT PERFORMANCE FOR GUARANTEE COMPLIANCE

OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= SEA LEVEL	C.G. STATION	= 107.0 IN. (FWD)
GROSS WEIGHT	= 2752.1 LB.	C_T	= 0.002753
ROTOR SPEED	= 354 RPM	CONFIGURATION	= CLEAN-ALL DOORS ON
			PARTICLE SEPARATOR REMOVED

NOTE: 1. SHP REQUIRED OBTAINED FROM FIGS. 23 THRU 25, APPENDIX VI.
2. SHP AVAILABLE OBTAINED FROM FIG 69, APPENDIX VI.



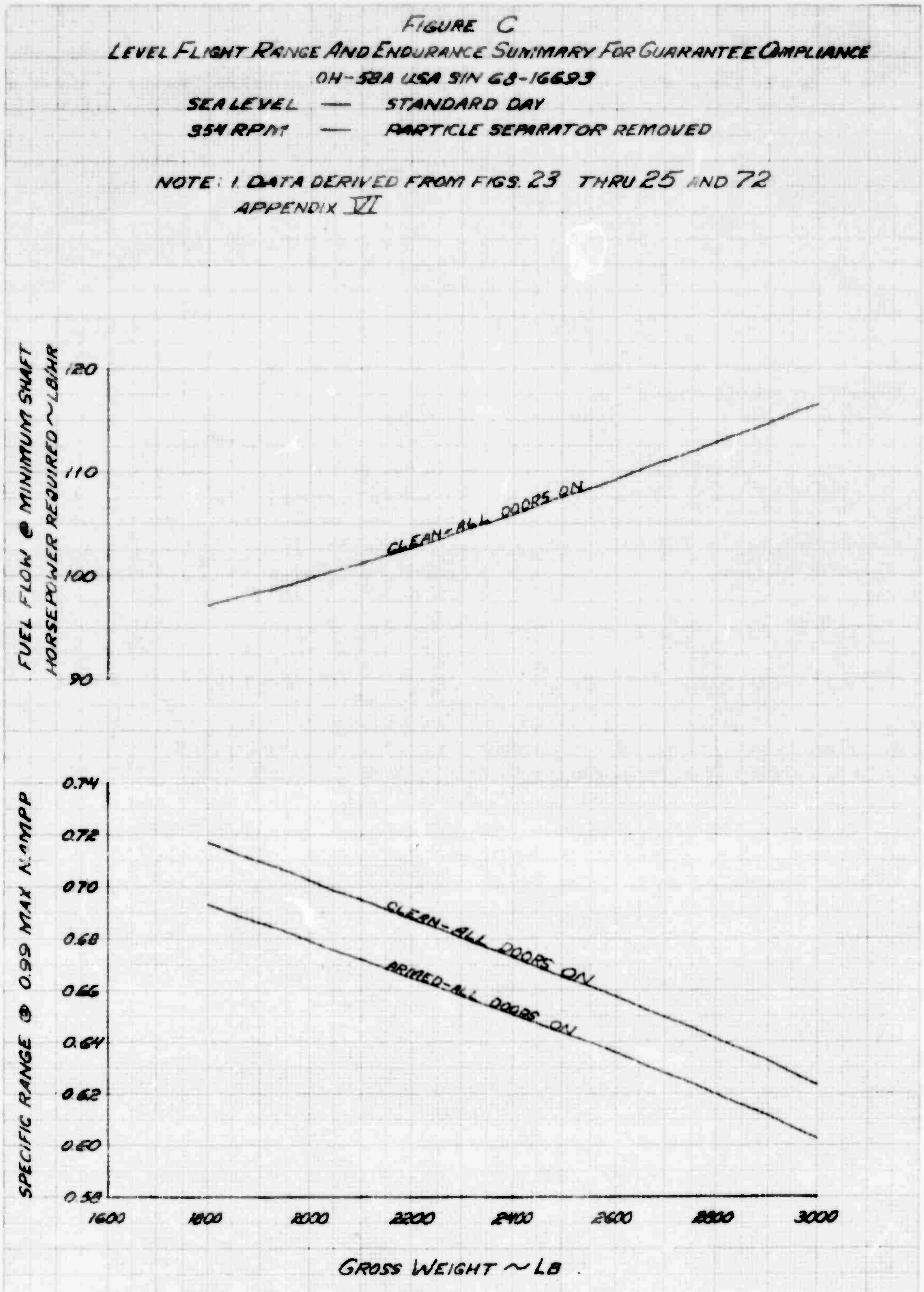


FIGURE C
LEVEL FLIGHT RANGE AND ENDURANCE SUMMARY FOR GUARANTEE COMPLIANCE
OH-58A USA S/N 68-16693
SEA LEVEL — STANDARD DAY
354 RPM — PARTICLE SEPARATOR REMOVED
NOTE: 1. DATA DERIVED FROM FIGS. 23 THRU 25 AND 72
APPENDIX VI
FUEL FLOW @ MINIMUM SHAFT HORSEPOWER REQUIRED ~ LB/HR
120
110
100
90
CLEAN-ALL DOORS ON
SPECIFIC RANGE @ 0.99 MAX NAMPP
0.74
0.72
0.70
0.68
0.66
0.64
0.62
0.60
0.58
CLEAN-ALL DOORS ON
ARMED-ALL DOORS ON
1600
1800
2000
2200
2400
2600
2800
3000
GROSS WEIGHT ~ LB

24. Using data from figure C, level-flight range guarantees for configurations I and II were computed to match the guaranteed values of 260 and 230 nautical air miles traveled, respectively. The range computations which define configuration I and II gross weights and were used to check the guarantees are summarized as follows:

Item	Configuration I[1] (lb)	Configuration II[2] (lb)
Basic mission weight without fuel	2308.2	2542.2
Fuel	443.9	417.8
Mission gross weight	2752.1	2960.0
Engine start gross weight	2752.1	2960.0
Warm-up (2 minute) at NRP[3]	-6.21	-6.21
Lift-off gross weight	[4]2745.89	[5]2953.79
Cruise fuel	[6]-393.3	[7]-369.81
Landing gross weight	[8]2352.59	[9]2583.98
Basic mission weight without fuel	-2308.2	-2542.2
Ten-percent fuel reserve	44.39	41.78

[1]Observation mission, unarmed, all doors on, sea level, 15°C, T63-A-700 engine, JP-4 fuel, no bleed air, anti-icer off, particle separator removed, 354 rpm, cruise at 0.99 maximum NAMPP specific range.
[2]Same as footnote 1 above except for: scout mission, armed with XM27E1, 2000 rounds of ammunition.
[3]Fuel flow = 186.3 lb/hr.
[4]Specific range = 0.6460 NAMPP; fuel flow = 111.8 lb/hr.
[5]Specific range = 0.6068 NAMPP.
[6]Fuel required for 260 nautical air miles.
[7]Fuel required for 230 nautical air miles.
[8]Specific range = 0.6762 NAMPP; fuel flow = 105.1 lb/hr.
[9]Specific range = 0.6372 NAMPP.

Formula:

$$\text{Range} = \text{Average NAMPP} \times \text{cruise fuel}$$

Configuration I Calculation:

$$\text{Range} = \frac{0.6460 + 0.6762}{2} \times 393.3$$

$$\text{Range} = 260.01 \text{ nautical miles}$$

Configuration II Calculation:

$$\text{Range} = \frac{0.6068 + 0.6372}{2} \times 369.81$$

$$\text{Range} = 230.02 \text{ nautical miles}$$

25. Using figure C and the range summary of configuration I, endurance was calculated to be 3.63 hours which exceeded the guarantee of 3.0 hours by 21 percent. The computation of endurance is summarized as follows:

Formula:

$$\text{Endurance} = \frac{\text{cruise fuel}}{\text{average fuel flow}}$$

Calculation:

$$\text{Endurance} = \frac{393.3}{\frac{111.8 + 105.1}{2}}$$

$$\text{Endurance} = 3.63 \text{ hours}$$

26. The results of the level-flight performance tests of the OH-58A with the particle separator installed were satisfactory. However, compressibility may decrease level flight performance significantly. Advancing tip Mach ($M_{N_{tip}}$), which is associated with compressibility, is mainly affected by temperature; that is, the colder the temperature, the higher the $M_{N_{tip}}$. One flight was conducted at a coefficient of thrust (C_T) of 0.003480, at a relatively cold temperature, and higher $M_{N_{tip}}$ to determine compressibility effect. The result is illustrated nondimensionally in figure D.

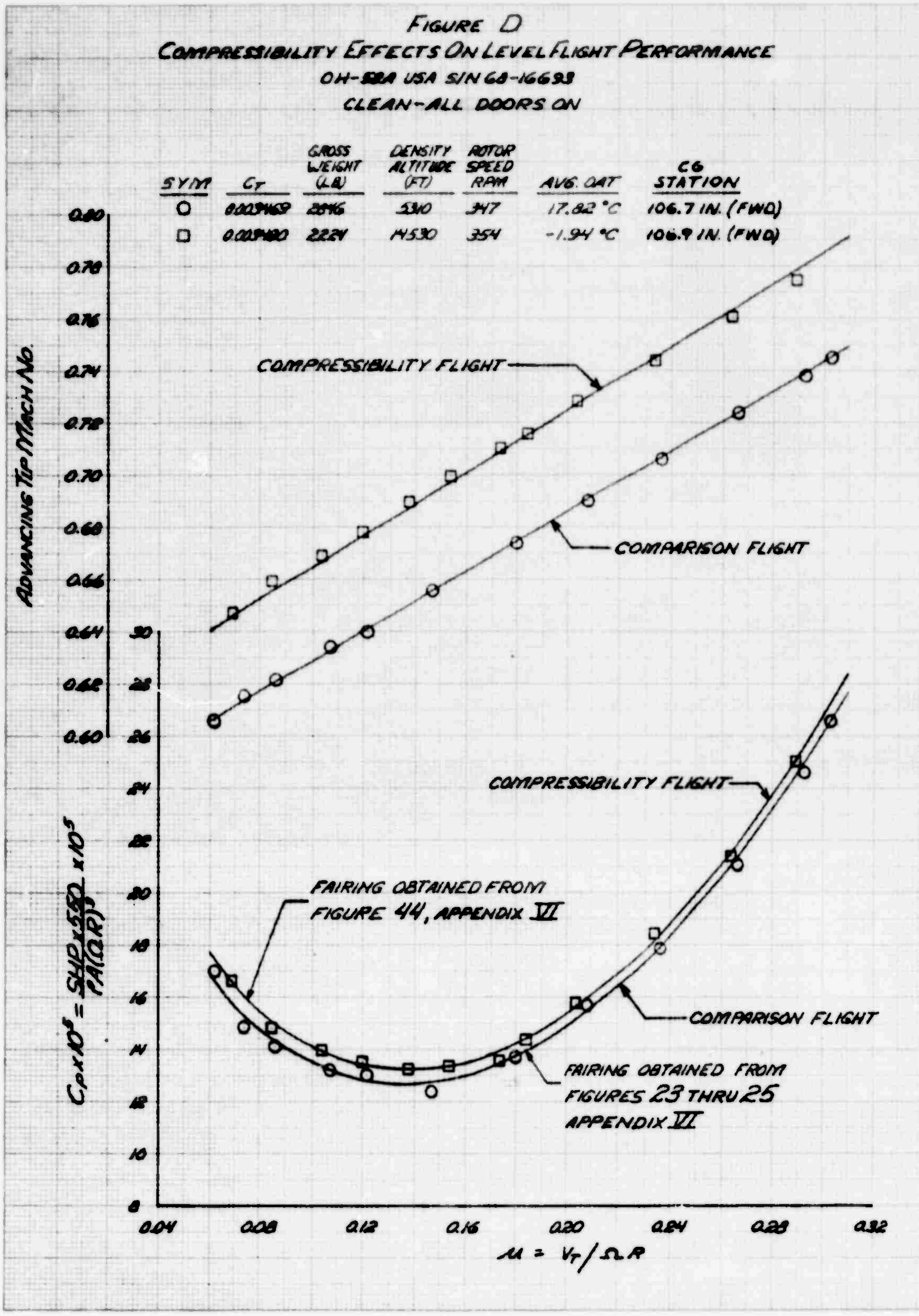


FIGURE D
COMPRESSIBILITY EFFECTS ON LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693
CLEAN-ALL DOORS ON
SYM | CT | GROSS WEIGHT (LB) | DENSITY ALTITUDE (FT) | ROTOR SPEED RPM | AVG. OAT | CG STATION
○ 0.003469 2846 5310 347 17.82 °C 106.7 IN. (FWD)
□ 0.003480 2224 14530 354 -1.94 °C 106.9 IN. (FWD)
ADVANCING TIP MACH NO.
COMPRESSIBILITY FLIGHT
COMPARISON FLIGHT
FAIRING OBTAINED FROM FIGURE 44, APPENDIX VI
FAIRING OBTAINED FROM FIGURES 23 THRU 25 APPENDIX VI
$C_P \times 10^5 = \frac{SHP \times 550}{\rho A(\Omega R)^3} \times 10^5$
$\mu = V_T / \Omega R$

27. Compared with another flight at a similar C_T as shown in figure D, the compressibility flight with the higher $M_{N_{tip}}$ resulted in an increased coefficient of power (C_P) at all airspeeds. Since the compared C_T's were not identical, further substantiation of compressibility may be found through a comparison of figures 23 through 25, appendix IV, with figure 44. Figure 44 indicates that an average increase of 3 shp is required at airspeeds between 26 and 115 KTAS. A more detailed investigation should be conducted at cold temperatures to determine compressibility effects on the performance capability of the OH-58A.

28. The unarmed OH-58A with all doors on was defined as the clean external configuration with zero equivalent flat plate area (F_e). Figure 29, appendix VI, illustrates that for each configuration the increased drag is uniquely constant at all C_T's for airspeeds between 20 and 120 KTAS. Compared with the clean external configuration, the armed helicopter with the XM27E1 armament system installed and all doors on resulted in an increased F_e of 1.2 square feet. Also, the armed configuration with the cargo doors off and all doors off showed a respective increase in F_e of 2.2 and 2.8 square feet. A range performance comparison among the various configurations at 354 rpm and configuration II gross weight (2960 pounds) at sea level, standard day is summarized in table 4.

Table 4. Range Summary[1] at Sea Level (15°C Day).

Configuration	Equivalent Flat Plate Area (ft^2)	Recommended Cruise[2] (KTAS)	Specific Range (NAMPP)
Unarmed, all doors on	0	104.5	0.603
Armed, all doors on	1.2	101.5	0.582
Armed, cargo doors off	2.2	99.7	0.567
Armed, all doors off	2.8	98.5	0.559

[1]Fuel flow based on engine model specification with particle separator installed.
[2]Airspeed based on high side of 0.99 maximum NAMPP.

29. With the exception of two flights, to determine cg effects all level-flight performance tests were conducted at a mission cg (FS 107.0). As may be seen in figure 30, appendix VI, at the forward cg limit (FS 105.2) the F_e increased 1.15 square feet, while at the aft limit (FS 114.2) the F_e decreased 0.22 square feet.

30. The increase in power required with various angles of sideslip is shown in figure 31, appendix VI, and is summarized in table 5.

Table 5. Difference in Shaft Horsepower Required Versus Sideslip.

Sideslip Angle (deg)	60 Knots True Airspeed (shp)	80 Knots True Airspeed (shp)	100 Knots True Airspeed (shp)	110 Knots True Airspeed (shp)
5 left	0.8	4.1	9.0	12.2
5 right	1.0	4.0	8.6	11.8
10 left	2.8	9.5	21.7	29.2
10 right	2.4	8.2	17.2	24.1
15 left	6.8	18.8	47.0	--
15 right	4.7	15.5	40.0	--

Autorotational Descent

31. Tests were conducted to determine the autorotational descent performance characteristics of the OH-58A. Test results were satisfactory and are presented in figures 54 through 57, appendix VI. To determine the airspeed for minimum rate of descent, a series of stabilized descents was conducted at each of the conditions listed in table 6. During each autorotational descent, rotor speed was held constant, and data were recorded at stabilized airspeeds from 30 to 80 KCAS in 10-knot increments. After the airspeed for minimum rate of descent was determined, another series of descents was conducted while maintaining constant airspeed and varying rpm from 330 to 390 in 10-rpm increments. The airspeed for maximum glide distance varied from 74.5 KCAS at 330 rpm to 81.5 KCAS at 390 rpm. The variation in horizontal glide distance over this rpm/airspeed range was less than 85 feet. It was, therefore, determined that for optimum performance an airspeed of from 74 to 78 KCAS and 360 rpm should be maintained during autorotational descents.

Table 6. Autorotational Descent Test Conditions.

Gross Weight (lb)	Rotor Speed (rpm)
2160	330, 354, 390
2640	330, 354, 390
2800	330, 354, 390

Power-on Landing

32. Satisfactory power-on landing performance was qualitatively evaluated in conjunction with the takeoff tests. The OH-58A is capable of accomplishing satisfactory landings under conditions where hover capability does not exist, and no special or unique pilot techniques or additional space is required. However, under such conditions, precise control manipulations are necessary to preclude touchdown with a high sink rate.

MISCELLANEOUS TESTS

Weight and Balance

33. The test helicopter was weighed before the start of the flight test program. The weighing was accomplished in a closed hangar with electrical load cells placed under the aircraft jack points. The basic weight (empty aircraft plus trapped fuel and oil) was 1555.0 pounds, and the cg location was at FS 118.5. This weight compared very well with the contractor weight on the same aircraft which was 1557.0 pounds. The specification guaranteed weight was 1586.2 pounds. After the instrumentation was installed, the test aircraft was reweighed using the same equipment and procedure, and a gross weight of 1676.5 pounds with a cg location at FS 115.8 was recorded. Throughout this evaluation, a mission cg location at FS 107 was used for all tests with the exception of two level-flight speed-power polars to determine cg effects.

Engine Starts

34. The OH-58A engine starting characteristics on the ground were satisfactory utilizing either the aircraft battery or external power units. Satisfactory engine starts were accomplished at field elevations ranging from sea level to 9500 feet under varying atmospheric conditions. No tendency toward "overtemp" was experienced during this test utilizing the procedures outlined in the operator's manual, TM 55-1520-228-10 (ref 5, app I).

35. Air starts were accomplished at pressure altitudes from 4000 to 11,500 feet. During this test, the gas producer speed (N_1) as well as the power turbine speed (N_2) were allowed to completely coast to a stop prior to initiating the restart. Five of seven start attempts were successfully accomplished without complications. All successful air starts were similar to the ground starts with the exception that the turbine outlet temperature (TOT) was observed to be slightly higher. One start attempt resulted in an abort because of overtemp conditions, and another abort was caused by insufficient gas producer rpm to effect light off. After each abort, a successful start was accomplished after descending to a lower altitude. One air start was satisfactorily accomplished with the boost pump inoperative at a pressure altitude of 10,100 feet. Altitude loss during the air start attempts was not excessive and averaged approximately 1200 feet from throttle closure to power application in the recovery. The same procedures that were used for ground starts produced satisfactory results during the air start tests.

Engine Characteristics

36. Tests were conducted to determine the compressor inlet pressure (P_{T_2}) and temperature (T_{T_2}) characteristics, both with and without the particle separator installed. In order to compare the installed test engine to an installed model specification engine, the calibrated test engine parameters of shaft horsepower, gas producer speed, turbine outlet temperature, and fuel flow were measured during the entire test program.

37. For the purpose of the engine inlet tests, an inlet ring with three pressure and three temperature probes was installed in the engine bellmouth. The probes were placed 120 degrees apart and were manifolded into one average reading for each parameter. The pressure probes were referenced to the test boom static source system to provide a differential pressure measurement. The temperature installation system measured temperature directly. Test results are presented in figures 59 and 60, appendix VI, as plots of compressor inlet temperature rise and compressor inlet pressure ratio (P_{T_2}/P_a) with the particle separator removed or installed versus calibrated airspeed. A comparison plot of contractor estimated losses and test data versus calibrated airspeed is presented in figure 58.

38. At zero airspeed, the particle-separator-removed configuration indicated a pressure loss of 0.3 percent and a temperature rise of 2°C while the installed configuration indicated a 1.5-percent pressure loss and a 2.4°C temperature rise. The takeoff power available at sea-level, standard-day conditions was reduced 7.5 shp by the installation of the particle separator. This represents a 2.4-percent power loss as compared to the maximum of 2.0 percent stipulated in the detail specification.

39. The detail specification estimated the compressor inlet temperature rise, at zero airspeed, to be 2.5°F for both the installed and removed particle separator conditions. Test data show a temperature rise of 4.32 and 3.60°F, respectively. The detail specification also estimated the compressor inlet pressure loss at zero airspeed to be 1.0 inches of water (H_2O) with the particle separator removed, and stipulated the loss with the particle separator installed to be no greater than 4.0 inches of water. Test data show losses of 1.31 and 6.12 inches of water, respectively. All contractual performance guarantees were met; however, the test data indicated inlet losses greater than those estimated with the particle separator removed and greater than the limits set forth in the detail specification for the particle-separator-installed condition. The magnitude of the inlet pressure loss, as compared with the detail specification requirement, was not sufficient to be called a shortcoming. Table 7 summarizes the comparison of both sources.

Table 7. Engine Inlet Losses.

Parameter	Particle Separator Removed		Particle Separator Installed	
	Estimated	Test Data	Estimated	Test Data
Pressure loss	1.0 in. of H_2O	1.31 in. of H_2O	4.0 in. of H_2O	6.12 in. of H_2O
Temperature rise	2.5°F	3.6°F	2.5°F	4.32°F

40. Figure 61, appendix VI, illustrates the exhaust pressure loss because of the exhaust extension installation. These data, used to predict power available and fuel flow for a specification engine, were furnished by the airframe contractor.

41. The referred terms of the engine parameters were used to compare the calibrated test engine with the model specification engine. Data on referred gas producer speed, shaft horsepower, and turbine outlet temperature are presented in figures 62 and 63, appendix VI. Fuel-flow data are not presented because a problem existed in the measurement of the fuel-flow rate which was attributed to the test instrumentation.

42. The referred shaft horsepower versus referred gas producer speed (fig. 62, app VI) shows that the gas producer speed for the test engine was below the model specification. At a gas producer speed equal to 100-percent rpm, the test engine produced 10 shp less than the model specification engine. These data indicate that the test engine may

have had a dirty compressor section. In these areas, the turbine outlet temperature and the trend of fuel-flow data show that the test engine performed better than the model specification engine.

Pitot-Static System Calibration

43. An airspeed calibration was performed with the test aircraft utilizing the ground speed course and trailing bomb methods. Test results are presented in figures 64 through 67, appendix VI.

Altimeter Calibration

44. The altimeter position error was calculated utilizing data collected during airspeed calibration tests. Test results are presented in figure 68, appendix VI.

Problems Encountered

45. During the conduct of this evaluation, the following problems were encountered:

a. Blow-by of oil from the main transmission oil filler cap when the system was properly serviced.

b. Failure of the tail rotor static stop rubber washer (P/N 206-010-777-1).

c. Failure of the linear actuator (N_2).

d. Malfunction of the fuel control (uncontrollable engine surge).

46. Prior to report compilation, information received from the Project Manager indicated that corrective action pertaining to these problems was being taken, and modification is being incorporated on current production aircraft.

CONCLUSIONS

GENERAL

47. Analysis of the test results obtained during this evaluation resulted in the following conclusions:

a. The overall performance characteristics of the OH-58A meet or exceed the detail specification guarantees (para 11).

b. The performance characteristics of the OH-58A are satisfactory for mission accomplishment under moderate temperature conditions (para 11).

c. At the configuration II gross weight, the OH-58A will not hover OGE on a 95°F hot day with the particle separator installed.

d. At the configuration II gross weight, the OH-58A cannot hover IGE on a 95°F hot day at pressure altitudes greater than 2450 feet.

e. Except for emergency tactical use, the OH-58A should not be flown at gross weights in excess of 2930 pounds at density altitudes higher than 2200 feet.

f. The compressor inlet losses with the particle separator installed are in excess of the detail specification requirement (para 39).

g. Compressibility significantly affects the performance of the OH-58A at high-altitude, low-temperature conditions (para 27).

DEFICIENCIES AND SHORTCOMINGS AFFECTING MISSION ACCOMPLISHMENT

48. During the conduct of this evaluation, no deficiencies or shortcomings were discovered.

RECOMMENDATIONS

49. As a result of this evaluation, the recommendations are as follows:

a. That the performance data generated during this evaluation be incorporated into the operator's manual.

b. That limitations be placed on the use of the OH-58A at combined high-gross-weight/high-ambient-temperature conditions.

c. That consideration be given to increasing the utility of the OH-58A by installing a more powerful engine.

d. That additional engineering tests be conducted to further define compressibility effects.

APPENDIX I. REFERENCES

1. Letter, AMCPM-LH-T, HQ, USAAVSCOM, 7 August 1968, subject: OH-58A Light Observation Helicopter Flight Test.

2. Detail Specification, Bell Helicopter Company, 206-947-031, *Light Observation Helicopter*, Revision Number R-4, 11 March 1969.

3. Test Plan, USAASTA, Project No. 68-30, *Engineering Flight Test of the Production OH-58A Helicopter, Unarmed and Armed with XM27E1 Weapons Subsystem*, May 1969.

4. Model Specification Number 803, Allison Division of General Motors Corporation, *Military Turboshaft Engine, Model T63-A-700*, 19 July 1967.

5. Operator's Manual, TM 55-1520-228-10, *Army Model OH-58A Helicopter*, 30 June 1969 with change 1 November 1969.

APPENDIX II. GENERAL AIRCRAFT INFORMATION

DIMENSIONS AND DESIGN DATA

Overall Dimensions

Aircraft length (rotor turning)	40 ft, 11.8 in.
Aircraft length (nose to tail skag)	32 ft, 2.0 in.
Width (rotor turning)	35 ft, 4 in.
Width (rotor static)	6 ft, 5.4 in.
Height (over main rotor blades at rest)	9 ft, 7.0 in.
Height (top of vertical stabilizer)	8 ft, 1.5 in.

Main Rotor

Number of blades	2
Diameter	35 ft, 4 in.
Blade chord (constant)	1.08 ft
Solidity	0.0390
Blade twist angle	-10.6 deg linear
Hub precone angle	3.0 deg
Airfoil section thickness	11.3%
Airfoil type	Modified "droop-snoot" airfoil

Tail Rotor

Number of blades	2
Diameter	5 ft, 2 in.
Blade chord	0.4375 ft
Blade twist angle	0 deg
Hub precone angle	0 deg
Airfoil section designation and thickness (constant)	NACA 0012.5

Control Travel

Cyclic stick (measured at center of grip):	
Longitudinal	6.0 in. fwd 6.0 in. aft
Lateral	5.15 in. right 5.15 in. left
Collective stick (measured at center of grip)	10.15 in.
Antitorque pedals (from neutral)	3.43 in. fwd 3.43 in. aft

Gear Ratio

Engine to main rotor	17.44:1
Engine to tail rotor	2.353:1

Operating Limitations

Power turbine speed (N_2)	101 to 103%
Turbine outlet temperature (TOT)	693°C (cont), 749°C (5 min)
Rotor rpm (power on)	347 to 354 rpm
Rotor rpm (power off)	330 to 390 rpm
Maximum airspeed (V_{max}), sea level	120 KIAS
Torque	79 psi (cont), 92 psi (max)

Powerplant

Aircraft power is provided by an Allison T63-A-700 free gas turbine engine which has a nominal rating of 270 shp at 100-percent N_2. As installed in the OH-58A, the engine is limited by either the output shaft torque or the gas producer turbine outlet temperature. For maximum continusous operation, these limits are 249 ft-lb torque (270 shp) at 6000 rpm or 693°C TOT, whichever is reached first. The takeoff power (maximum of 5 minutes continuous operation) limits are 293 ft-lb torque (317 shp) or 749°C. The engine consists of a multistage axial-centrifugal flow compressor, a single combustion chamber, a two-stage gas producer turbine and a two-stage power turbine which supplies the output power of the engine.

Fuel System

The helicopter fuel system incorporates a single-bladder type, self-sealing fuel cell with a total usable capacity of 73 United States gallons. The cell is located below and aft of the passenger seat. Mounted in the bottom of the cell is one boost pump, one fuel quantity transmitter, one low fuel transmitter and one fuel sump drain and defuel valve. Installed in the top of the cell is one fuel quantity transmitter, a vent line, a boost pump pressure switch and a governor return line. A fuel filler cap is located on the right side, just aft of the passenger door. The fuel shut-off valve is mounted on the right side of the aircraft above the fuel cell cavity and is manually operated.

Electrical System

The OH-58A electrical systems consist of a 28-volt, direct current (DC) dual bus system and a 115-volt 400 Hertz alternating current (AC) system.

The DC system is normally powered by a vented 24-volt, 13-ampere-hour, nickel-cadmium battery and a starter generator. The starter generator is used to start the aircraft engine, recharge the battery and provide primary 28-volt DC power for the aircraft electrical system. During ground operations, external DC power may be connected to the aircraft through a polarized, external power receptacle located on the right side of the fuselage below the baggage compartment.

The alternating current system is powered by a 65 volt-ampere, solid state inverter. The inverter delivers 115-volt AC, 400 Hertz to the AC bus. AC power is used to energize the attitude gyro and gyro compass.

APPENDIX III. DATA REDUCTION METHODS

INTRODUCTION

1. This appendix contains the formulas used to calculate the OH-58A performance capabilities. These formulas correct test-day conditions to standard-day conditions and also provide the necessary tools with which to predict helicopter performance for various atmospheric conditions. The following performance parameters are discussed.

a. Airspeed position error.

(1) Ground speed course method.

(2) Trailing bomb method.

b. Altimeter position error.

c. Shaft horsepower required.

d. Compressor inlet characteristics.

e. Hover.

f. Takeoff.

g. Climb.

h. Level flight and specific range.

i. Autorotation.

GENERAL

2. The basic nondimensional, helicopter equations that were used for hover, takeoff, climb, and level-flight analyses are defined as follows:

a. Coefficient of power (C_P):

$$C_P = \frac{SHP \times 550}{\rho A(\Omega R)^3} \qquad (1)$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{W}{\rho A(\Omega R)^2} \qquad (2)$$

c. Airspeed ratio (μ):

$$\mu = \frac{V_T}{\Omega R} \qquad (3)$$

where: SHP = Shaft horsepower

550 = Conversion factor (ft-lb/sec per shp)

ρ = Density (slugs/ft^3)

A = Main rotor disc area (ft^2)

Ω = Main rotor angular velocity (radians/sec)

R = Main rotor radius (ft)

W = Aircraft gross weight (lb)

V_T = True airspeed (kt)

Airspeed Position Error

3. The airspeed position error was determined by two methods:

a. Ground speed course method:

$$\Delta V_{PC} = V_{cal_{std}} - V_{IC_{test}} \tag{4}$$

$$\Delta V_{PC} = \left[\frac{(S \div t)_1 + (S \div t)_2}{2\ (1.6889)} \times \sqrt{\sigma}_{avg}\right] - \frac{V_{IC_1} + V_{IC_2}}{2} \tag{5}$$

where: ΔV_{PC} = Airspeed position error (kt)

$V_{cal_{std}}$ = Standard calibrated airspeed obtained from the ground speed course (kt)

$V_{IC_{test}}$ = Test system indicated airspeed corrected for instrument error (kt)

S = Course length (ft)

t = Time required to travel the course distance (sec)

$\sqrt{\sigma}_{avg}$ = Average density ratio at the average density altitude

Subscripts 1 and 2 = Reciprocal headings

1.6889 = Conversion factor (ft/sec per kt)

b. Trailing bomb method:

$$\Delta V_{PC} = V_{cal_{bomb}} - V_{IC_{test}} \tag{6}$$

where: ΔV_{PC} = Airspeed position error (kt)

$V_{cal_{bomb}}$ = Bomb system indicated airspeed corrected for instrument error (kt)

Altitude Position Error

4. The altitude position error for the standard ship system was calculated by using the airspeed position error from the ship's system.

$$\Delta H_{PC} = \left\{ \frac{58.566}{\sigma_s} \left(\frac{V_{IC}}{a_{SL}} \right) \left[1 + 0.2 \left(\frac{V_{IC}}{a_{SL}} \right)^2 \right]^{2.5} \right\} \Delta V_{PC} \tag{7}$$

where: ΔH_{PC} = Altitude position error (ft)

58.566 = Conversion factor

σ_s = Standard-day air density ratio at the test indicated altitude

V_{IC} = Test system indicated airspeed corrected for instrument error (kt)

a_{SL} = 661.48 (kt)

ΔV_{PC} = Airspeed position error (kt) (4, 5, 6)

Shaft Horsepower Required

5. The shaft horsepower required was determined by the following relationship:

$$SHP = \frac{2\pi \times K_t \times GR \times N_R \times TRQ}{33{,}000} \tag{8}$$

where: SHP = Shaft horsepower

K_t = Conversion factor to change measured engine torque pressure (psi) to ft-lb

GR = Gear ratio of the output shaft rotational speed to the main rotor rotational speed

N_R = Main rotor speed (rpm)

TRQ = Engine torque pressure (psi)

33,000 = Conversion factor (ft-lb/min per shp)

Compressor Inlet Characteristics

6. The compressor inlet temperature and pressure characteristics were determined by the following formulas:

a. Temperature rise:

$$\Delta T = CIT_{ic} - T_{a_{ic}} \tag{9}$$

where: ΔT = Temperature difference (°C)

CIT_{ic} = Indicated compressor inlet total temperature corrected for instrument error (°C)

$T_{a_{ic}}$ = Indicated ambient temperature corrected for instrument error (°C)

b. Pressure ratio:

$$\frac{P_{T_2}}{P_a} = \frac{\Delta CIP_{ic}}{P_{a_c}} + 1 \tag{10}$$

where: P_{T_2} = Compressor inlet total pressure (in. of Hg)

P_a = Ambient total pressure (in. of Hg)

ΔCIP_{ic} = Indicated compressor inlet pressure difference ($P_{T_2} - P_a$) corrected for instrument error (in. of Hg)

P_{a_c} = P_a at the indicated pressure altitude plus ΔP_a for altitude position error using formula 7 (in. of Hg)

Hover

7. Hover performance was determined in ground effect (IGE) and out of ground effect (OGE) by the free-flight hover technique. Formulas 1 and 2 were used to define the hover capability.

Takeoff

8. Formulas 1, 2, and 11 were used to determine the takeoff performance.

$$\Delta C_P = \left(SHP_{avail} - SHP_{req} \text{ at 2 ft} \right) \left(\frac{550}{\rho A (\Omega R)^3} \right) \qquad (11)$$

where: ΔC_P = Nondimensional factor for excess power (for each ΔC_P, a plot was constructed to relate the distance required to clear a 50-foot obstacle and the selected climb-out airspeed over the obstacle)

SHP_{avail} = Shaft horsepower available for the installed test engine at takeoff atmospheric conditions

SHP_{req} at 2 ft = Shaft horsepower required for a 2-foot hover skid height at takeoff atmospheric conditions

550 = Conversion factor (ft-lb/sec per shp)

Climb

9. The climb schedules used during this test program were determined by the airspeed for minimum power required in level flight. All climbs were flown with pressure altitude as the reference. Sawtooth climbs were flown to determine the coefficient of power correction (K_p) and the coefficient of weight correction (K_w). K_p and K_w were used to solve for the difference in rate of climb caused by differences in shaft horsepower and gross weight, respectively. These differences occur when the performance of an installed test engine is corrected to a model specification engine for standard-day conditions.

$$\Delta R/C_p = K_p \times \frac{\Delta SHP}{GW_t} \times 33{,}000 \qquad (12)$$

$$\Delta R/C_w = K_w \times SHP_s \times 33{,}000 \left(\frac{1}{GW_s} - \frac{1}{GW_t}\right) \qquad (13)$$

where: $\Delta R/C_p$ = Rate of climb difference due to power difference (ft/min)

K_p = Coefficient of power correction

ΔSHP = Difference in standard shaft horsepower available and test shaft horsepower measured

GW_t = Test gross weight (lb)

K_w = Coefficient of weight correction

SHP_s = Standard shaft horsepower obtained from a model specification engine

GW_s = Standard gross weight (lb)

$\Delta R/C_w$ = Rate of climb difference due to weight difference (ft/min)

10. The observed rate of climb was corrected to tapeline rate of climb by the equation:

$$R/C_T = \frac{dhp}{dt} \times \frac{T_t}{T_s} \qquad (14)$$

where: R/C_T = Tapeline rate of climb (ft/min)

$\frac{dhp}{dt}$ = Slope of pressure altitude versus time curve at a given pressure altitude (ft/min)

T_t = Test ambient air temperature at the pressure altitude at which the slope is taken (°K)

T_s = Standard ambient air temperature at the pressure altitude at which the slope is taken (°K)

11. The standard rate of climb was finally determined by the summarized equation:

$$R/C_s = R/C_T + \Delta R/C_p + \Delta R/C_w \qquad (15)$$

where: R/C_s = Final rate of climb standard (ft/min)

R/C_T = Tapeline rate of climb (ft/min)

$\Delta R/C_p$ = Rate of climb difference due to power difference (ft/min)

$\Delta R/C_w$ = Rate of climb difference due to weight difference (ft/min)

Level Flight and Specific Range

12. Level-flight speed-power performance was determined by using equations 1, 2, and 3. Each speed power was flown at a predetermined C_T with rotor speed held constant. To maintain W/ρ approximately constant, altitude was increased as fuel was consumed.

13. Test-day level-flight power was corrected to standard-day conditions by assuming that the test-day dimensionless parameters, Cp_t, C_{T_t}, and μ_t, are independent of atmospheric conditions. Consequently, the standard-day dimensionless parameters, Cp_s, C_{T_s}, and μ_s, are identical to Cp_t, C_{T_t}, and μ_t, respectively. A corrollary to the above assumption relates:

$$\rho_s = \rho_t\left(\frac{W_s}{W_t}\right) \qquad (16)$$

where: ρ = Density (slugs/ft^3)

W = Gross weight (lb)

Subscript t = Test day

Subscript s = Standard day

14. Equation 16 defines the standard-day density (ρ_s) which is required for presentation of test-day data at a standard gross weight (W_s).

15. From the definition of C_P (equation 1), the following relationship can be derived:

$$SHP_s = SHP_t \times \frac{\rho_s}{\rho_t} \qquad (17)$$

16. The relationship shown by equation 17 then defines the standard-day power required for flying at the same thrust, power, and airspeed coefficients as on the test day but under standard-day conditions. Each level-flight speed-power point was corrected in this fashion to standard-day conditions at the target gross weight.

17. Specific range was calculated using the nondimensional level-flight performance curve and the specification fuel-flow characteristics:

$$NAMPP = \frac{V_T}{W_f} \qquad (18)$$

where: NAMPP = Nautical air miles per pound of fuel (naut mi/lb)

V_T = True airspeed (kt)

W_f = Fuel flow (lb/hr)

<u>Autorotation</u>

18. The autorotational performance was determined using equation 15 except that R/C_T was re-defined as R/D_T.

$$R/D_T = \frac{dhp}{dt} \times \frac{T_t}{T_s} \qquad (19)$$

where: R/D_T = Tapeline rate of descent (ft/min)

APPENDIX IV. TEST INSTRUMENTATION

GENERAL

The test instrumentation used during this evaluation was supplied, installed and maintained by USAASTA. Sensitive instruments were calibrated to record the following parameters:

Pilot/Engineer Panel

Airspeed (boom system)
Altitude (boom system)
Airspeed (standard system)
Altitude (standard system)
Outside air temperature
Rate of climb
Rotor speed
Angle of sideslip
Fuel counter
Gas producer speed (N_1)
Torquemeter oil pressure
Turbine outlet temperature (T_{t_5})
Compressor inlet total temperature
Compressor inlet total pressure
Collective position indicator
Photopanel frame counter
Stepper motor indicator

Photopanel

Airspeed (boom)
Altitude (boom)
Outside air temperature
Rotor speed
Fuel counter
Time of day
Gas producer speed (N_1)
Torquemeter oil pressure
Turbine outlet temperature (T_{t_5})
Photopanel frame counter
Stopwatch

APPENDIX V. PHOTOGRAPHS

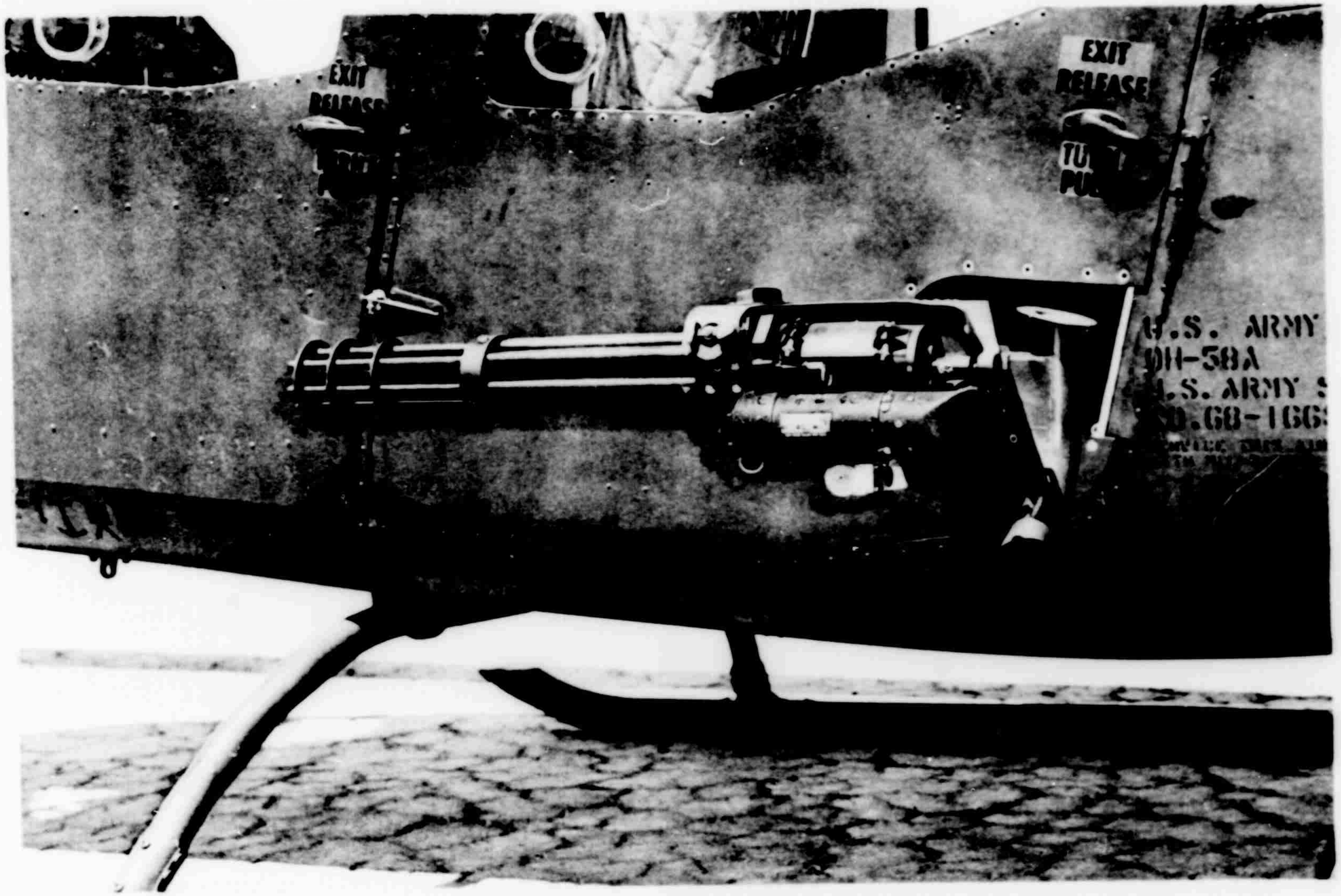


Photo 1. XM27E1 Armament System.

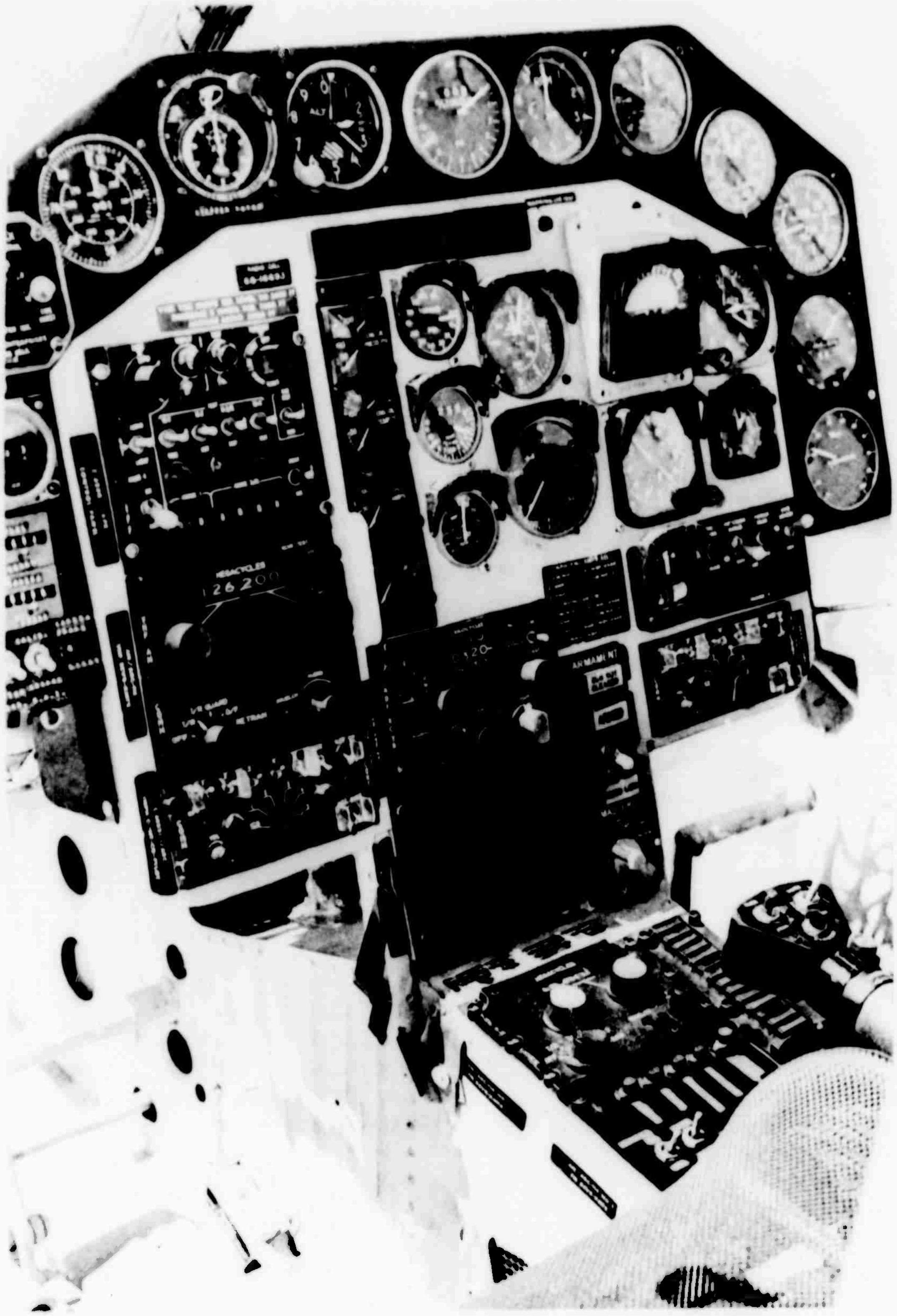


Photo 2. Cockpit Instrumentation.

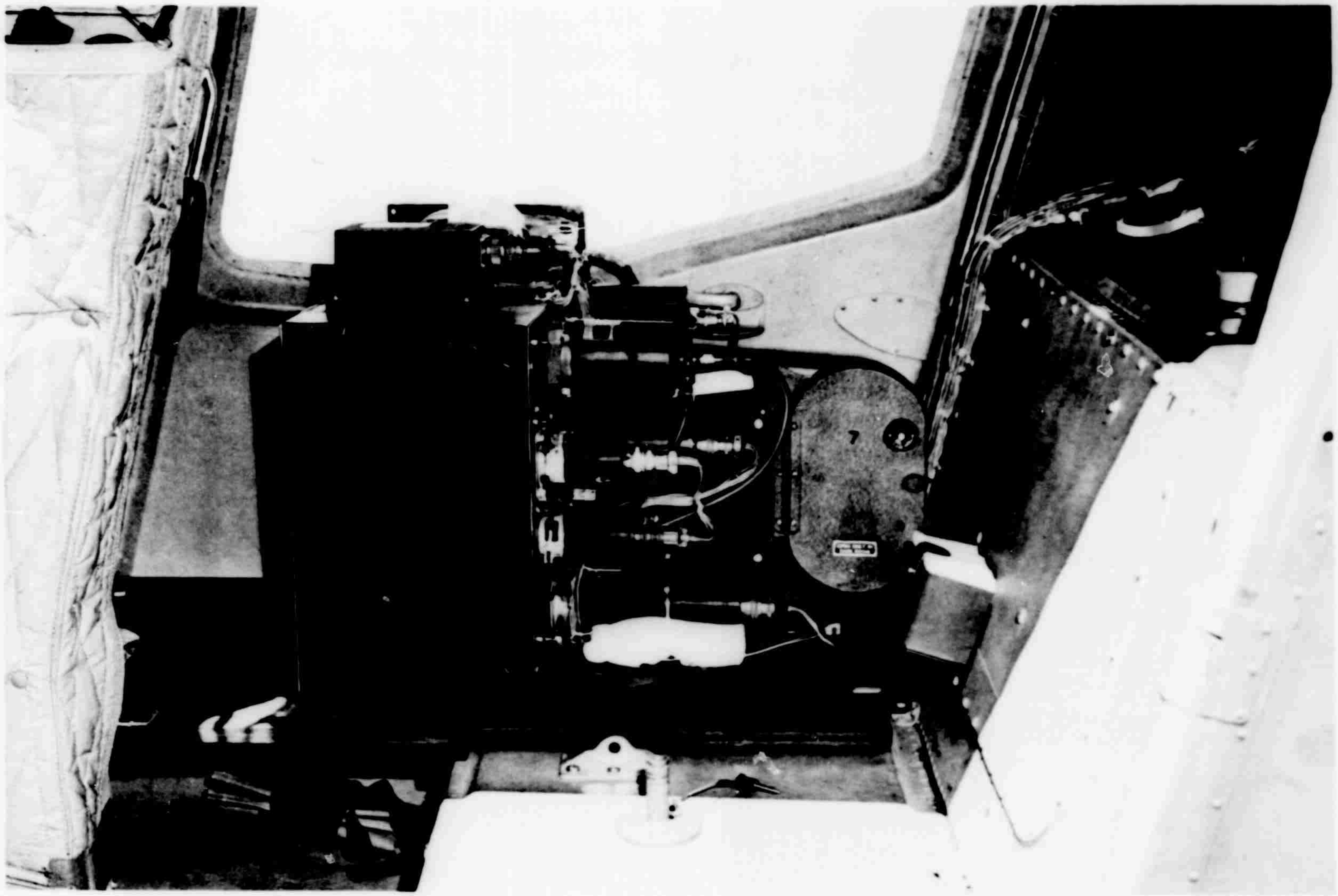


Photo 3. Photopanel Location.

APPENDIX VI. TEST DATA

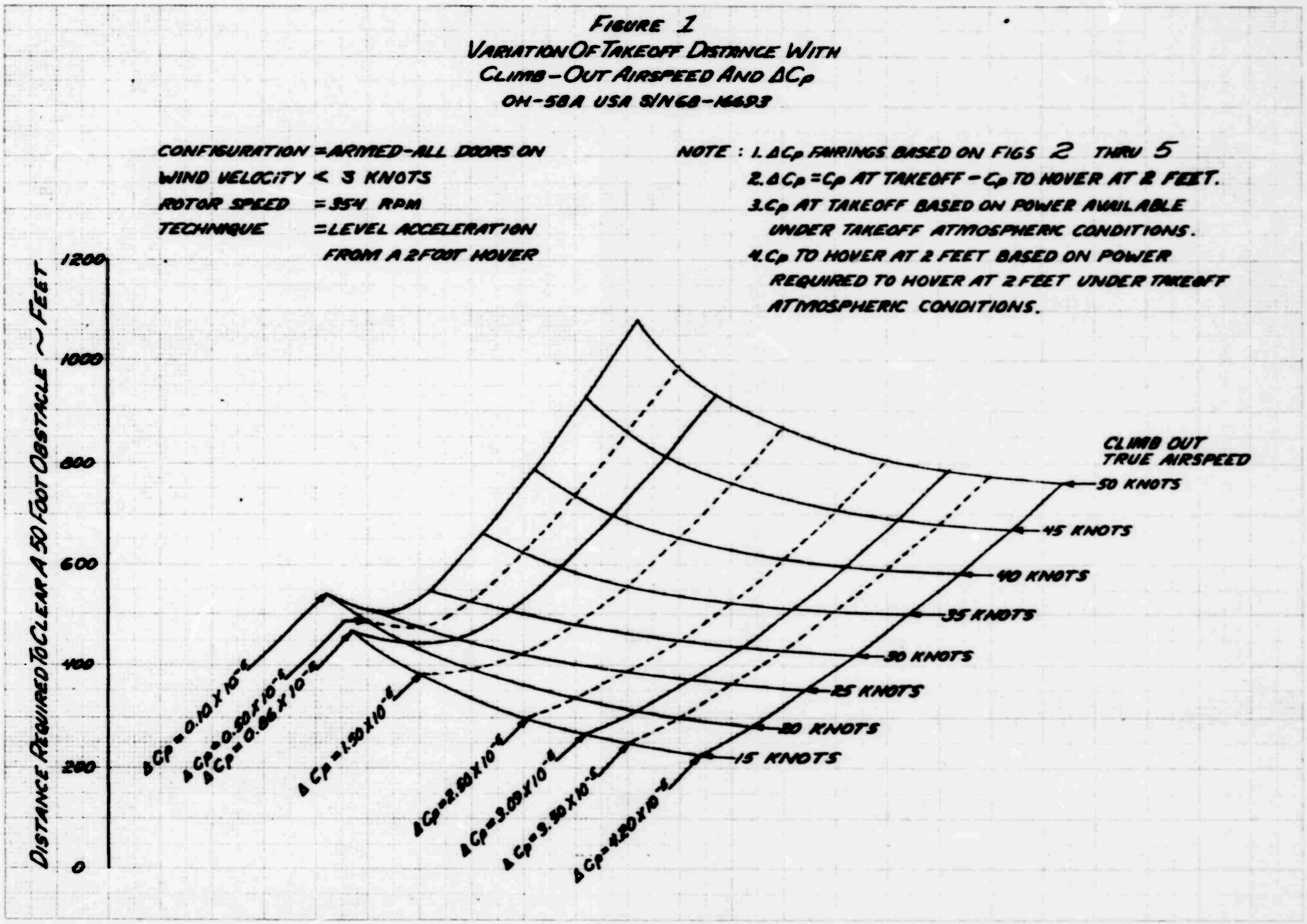


FIGURE 1
VARIATION OF TAKEOFF DISTANCE WITH CLIMB-OUT AIRSPEED AND ΔC_p
OH-58A USA S/N 68-16693

FIGURE 2
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FT OBSTACLE
OH-58A USA S/N 68-16693

ROTOR RPM	= 354	WIND VELOCITY	< 3 KTS
GROSS WEIGHT	= 2965 LB	FREE AIR TEMP	= 11.9 °C
PRESSURE ALTITUDE	= 4150 FT	ΔC_p	= 4.2×10^{-6}
TECHNIQUE	= LEVEL ACCELERATION FROM A 2 FT HOVER.	CONFIGURATION	= ARMED-ALL DOORS ON

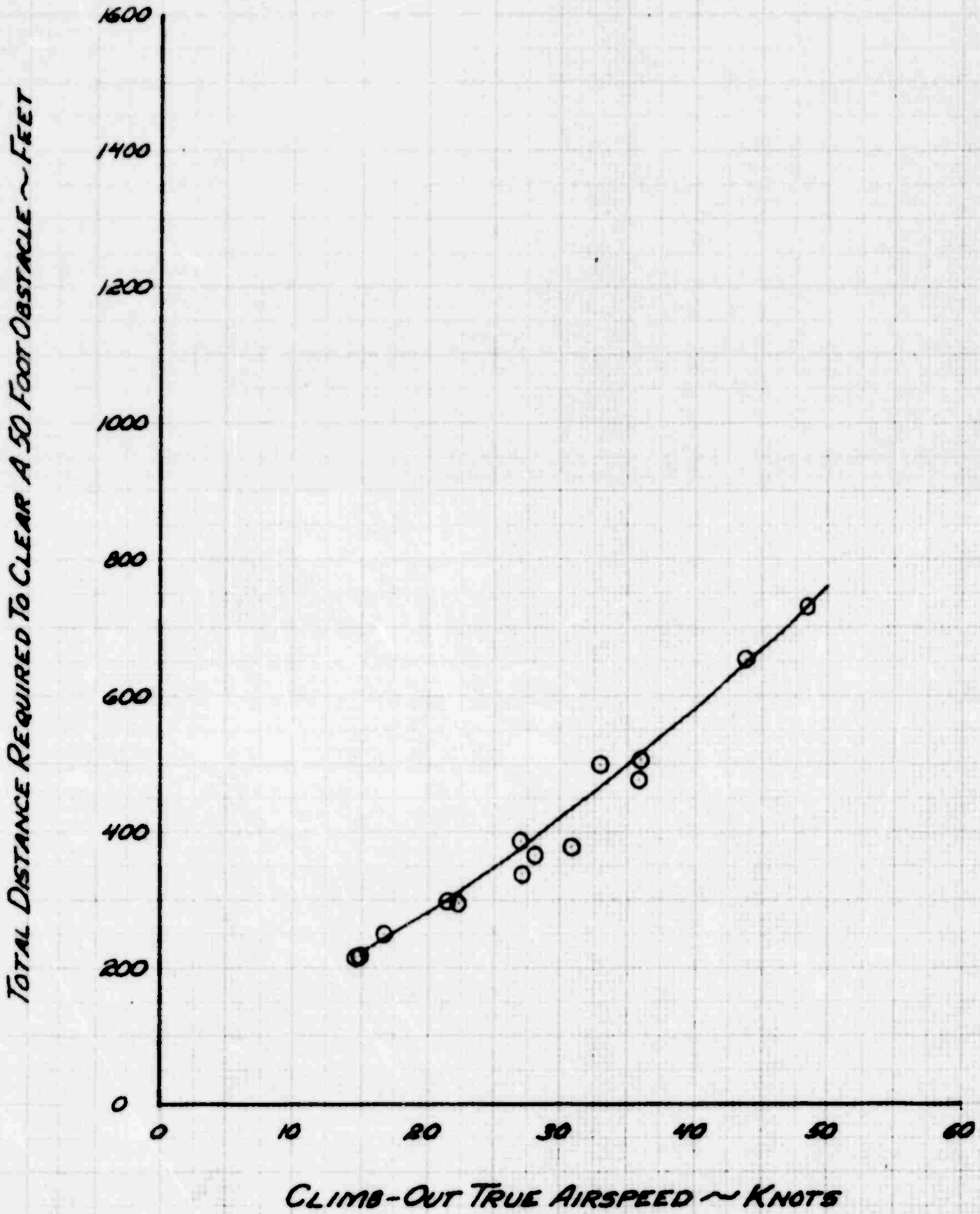


FIGURE 3
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FT. OBSTACLE
OH-58A USA S/N 68-16693

ROTOR RPM = 354
GROSS WEIGHT = 3030 LB
PRESSURE ALTITUDE = 4075 FT
TECHNIQUE = LEVEL ACCELERATION FROM A 2 FT. HOVER

WIND VELOCITY < 3 KTS
FREE AIR TEMP = 14.7°C
ΔC_P = 3.09×10^{-6}
CONFIGURATION = ARMED-ALL DOORS ON

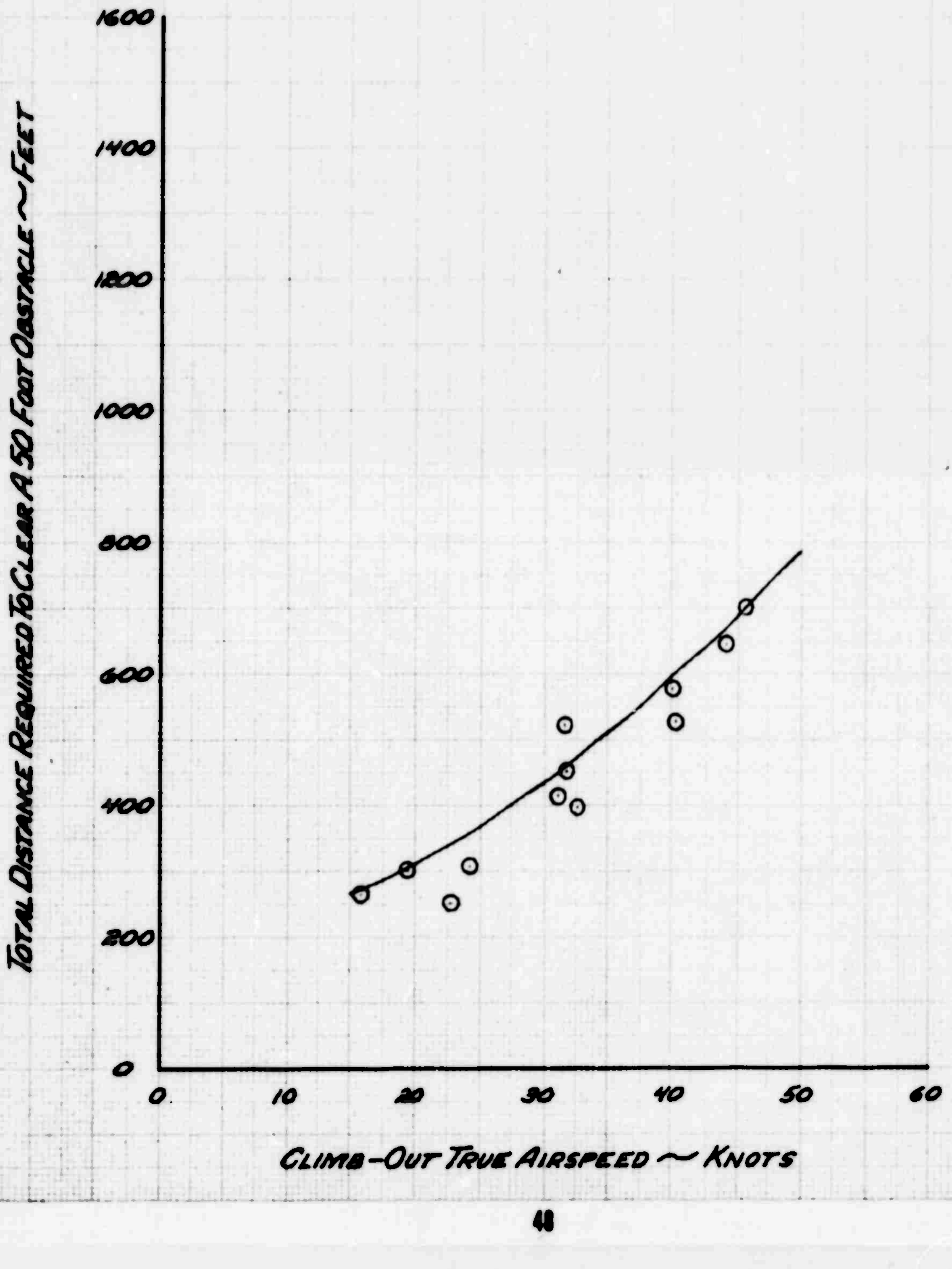
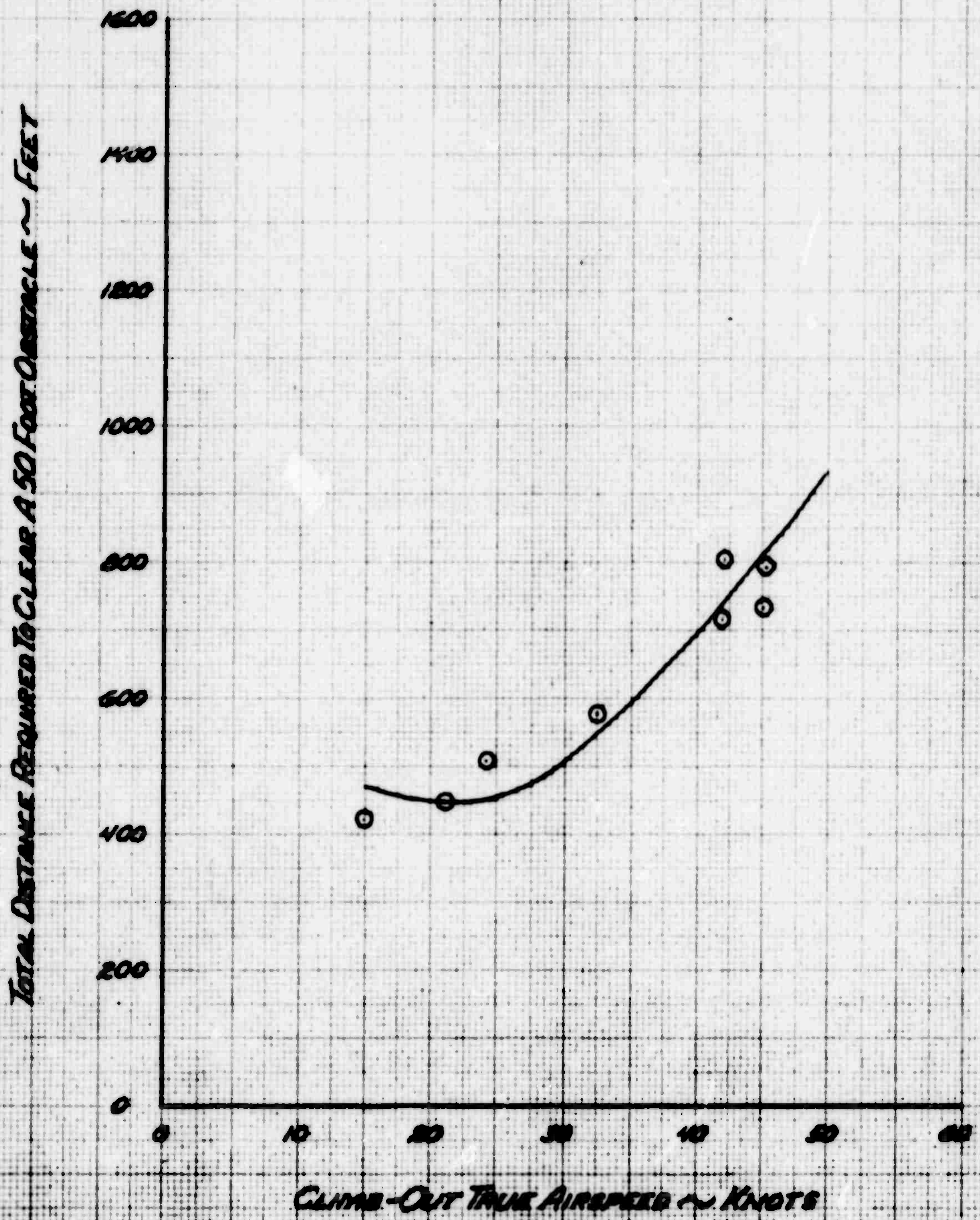


FIGURE 4
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FT. OBSTACLE
OH-58A USA S/N 68-16693

ROTOR RPM	= 354	WIND VELOCITY	< 3 KTS
GROSS WEIGHT	= 2769 LB	FREE AIR TEMP	= 0.3 °C
PRESS ALTITUDE	= 8950 FT	ΔC_P	= 0.86×10^{-4}
TECHNIQUE	= LEVEL ACCELERATION FROM A 2 FT. HOVER	CONFIGURATION	= ARMED-ALL DOORS ON



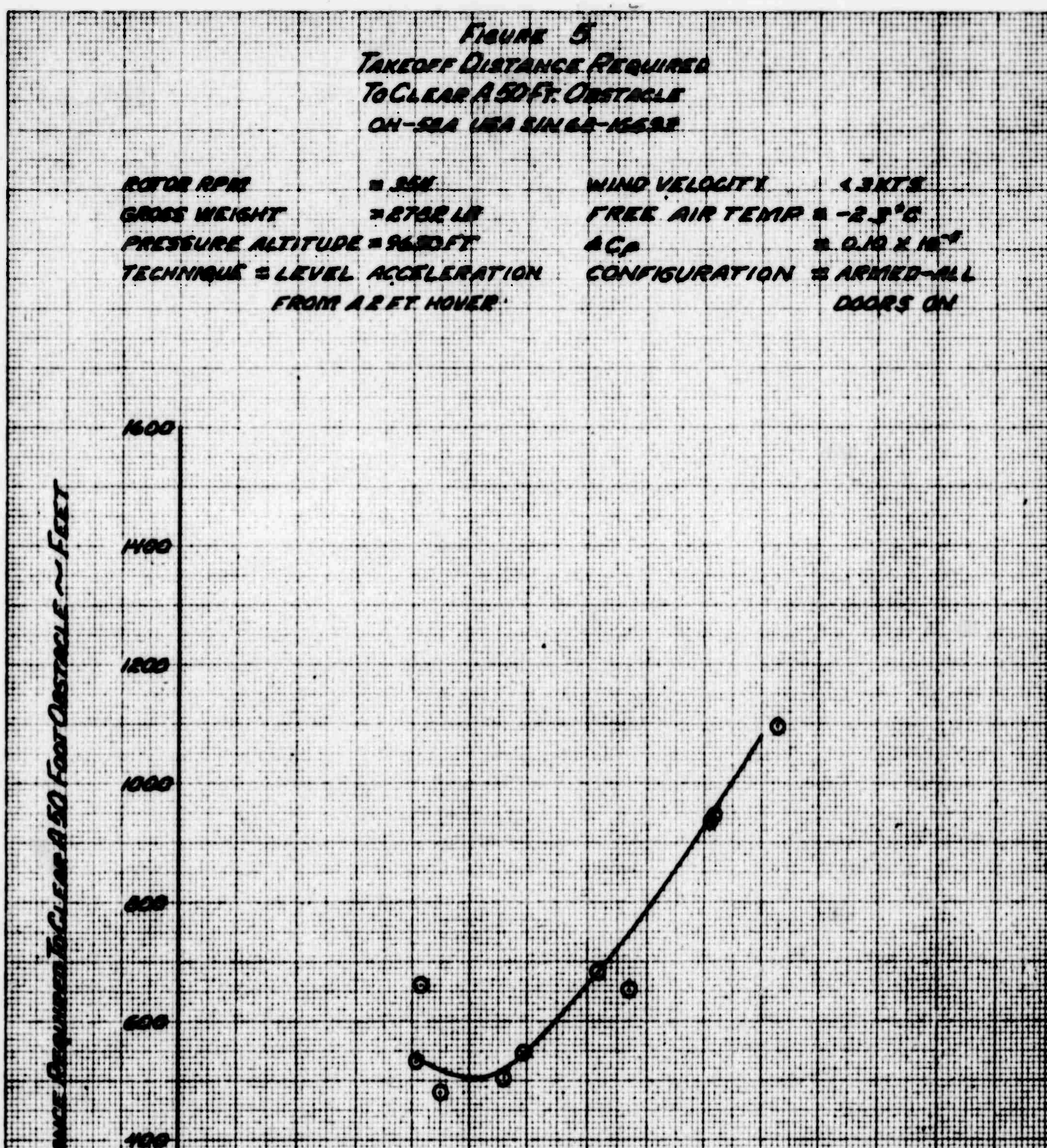


FIGURE 5
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FT. OBSTACLE
OH-58A USA S/N 68-16693

FIGURE 6

SUMMARY HOVERING PERFORMANCE

OH-58A USA S/N 68-16693

T63-A-700 S/N 402811

FREE FLIGHT METHOD

ROTOR SPEED = 354 RPM

WINDS LESS THAN 3 KNOTS

TAKE-OFF POWER

NOTE: CURVES DERIVED FROM FIGS. 8 THRU 14 AND 70

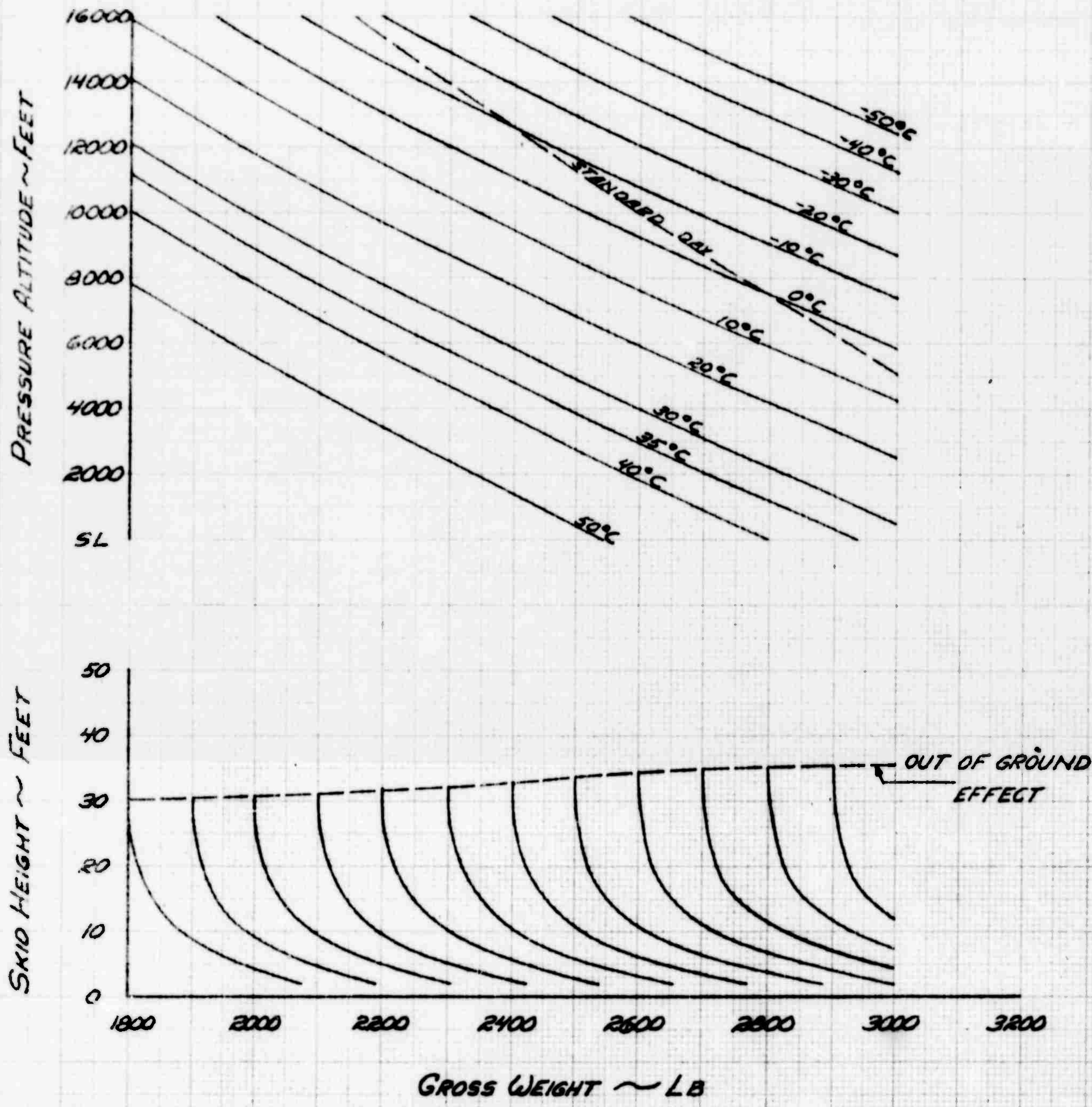


FIGURE 7
SUMMARY HOVERING PERFORMANCE
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811
FREE FLIGHT METHOD
ROTOR SPEED = 354 RPM
WINDS LESS THAN 3 KNOTS
NORMAL RATED POWER

NOTE: CURVES DERIVED FROM FIGS. 8 THRU 14 AND 71

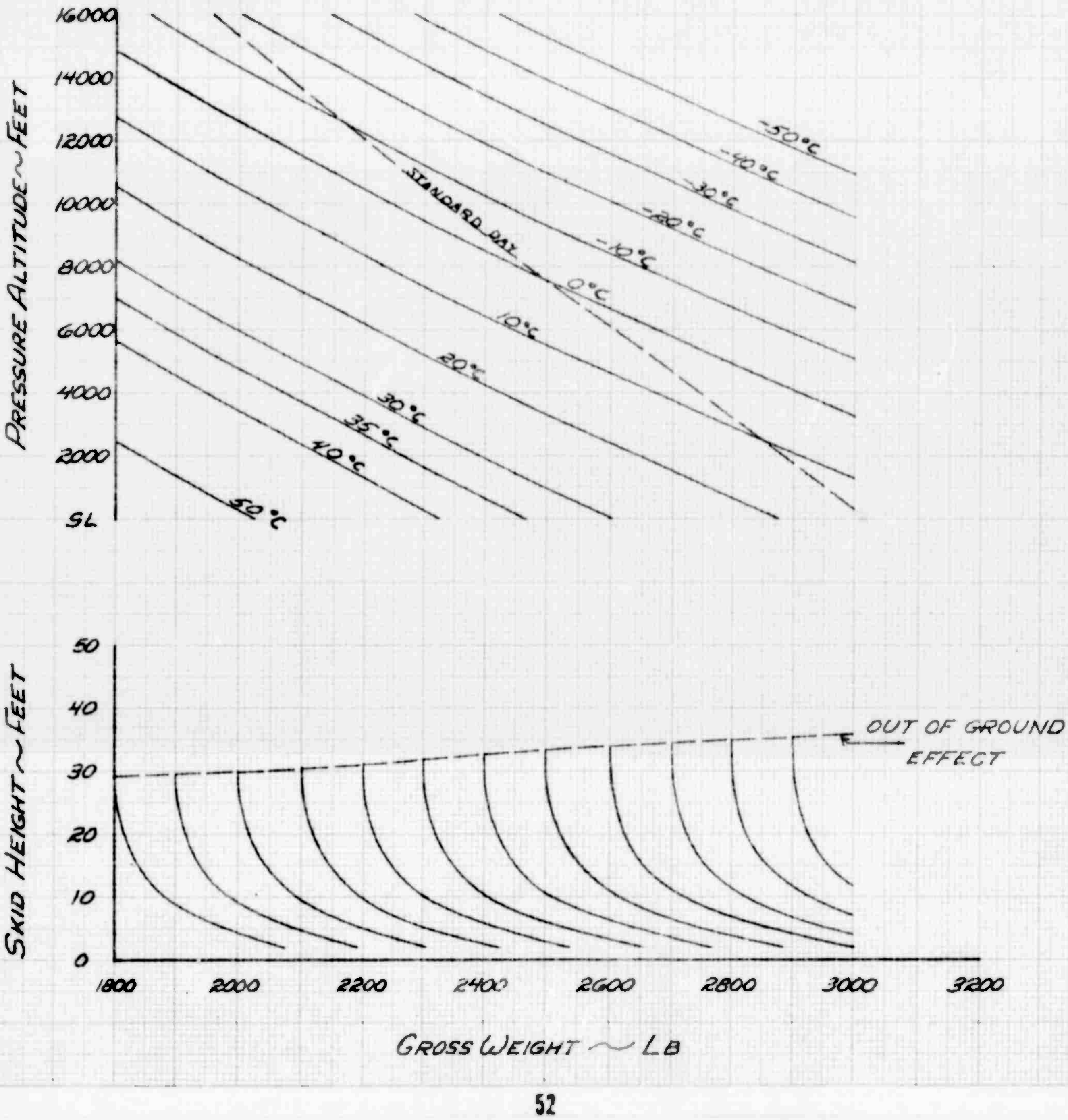


FIGURE 8
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS

NOTE
1. DERIVED FROM FAIRED CURVES ON FIGS 9 THRU 14
2. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET.

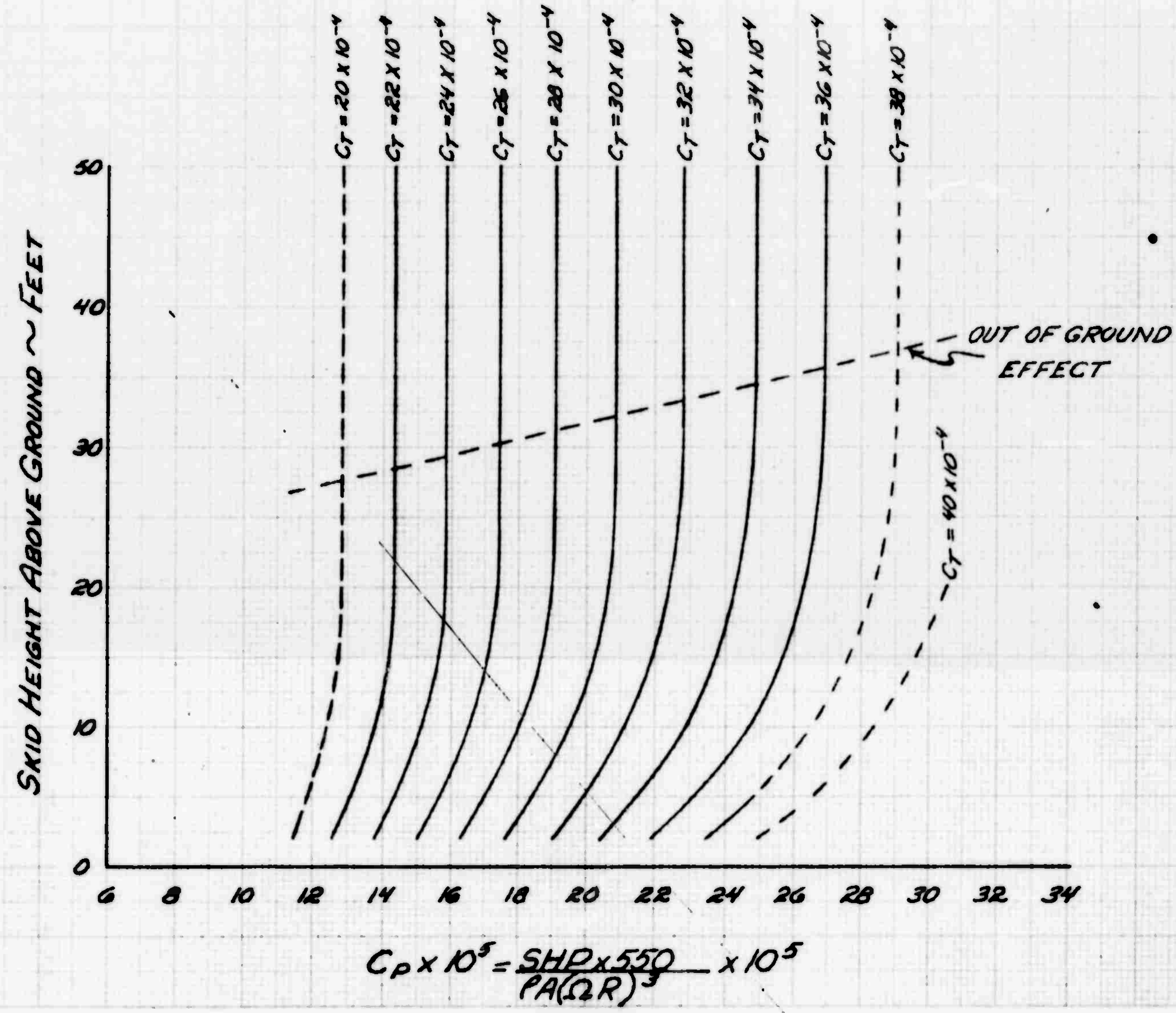


FIGURE 9

NON-DIMENSIONAL HOVERING PERFORMANCE

OH-58A USA S/N68-16693

T63-A-700 S/N402811

FREE FLIGHT METHOD

WINDS LESS THAN 3 KNOTS

NOTE: 1. CURVES OBTAINED FROM FIGS. 10 THRU 14

2. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET.

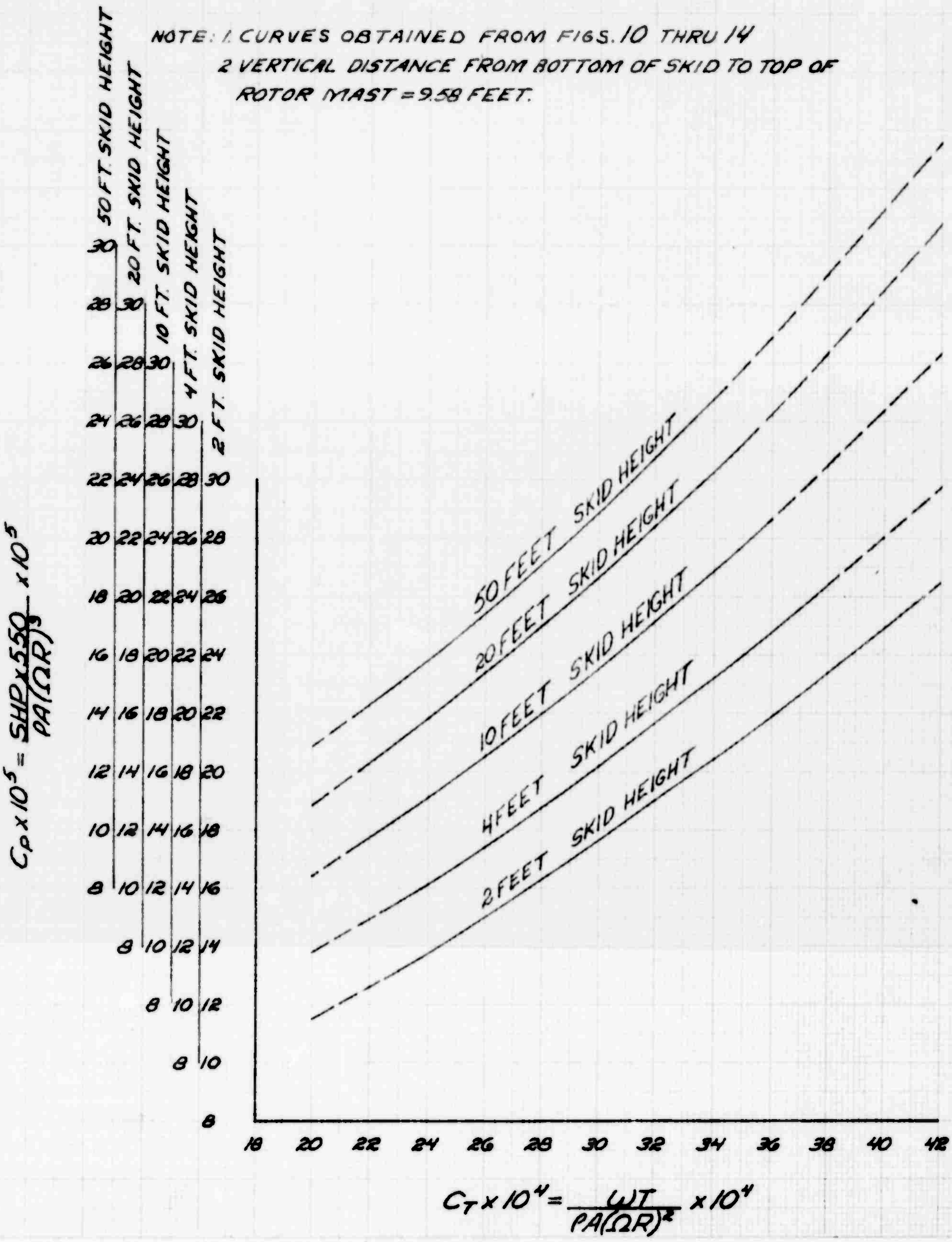


FIGURE 10
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811
SKID HEIGHT = 2 FEET
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS

SYM	GROSS WEIGHT (LB)	CONFIG	SYM	GROSS WEIGHT (LB)	CONFIG
○	3010	CLEAN	✧	2620	CLEAN
□	2870	CLEAN	☆	2535	CLEAN
◇	2795	ARMED	▲	2515	ARMED
△	2765	CLEAN	⬭	2340	CLEAN
▽	2765	ARMED	0	2195	CLEAN
○	2715	CLEAN			

NOTE:
1. UNSHADED SYMBOLS DENOTE 347 RPM.
2. SHADED SYMBOLS DENOTE 354 RPM.
3. UNFLAGGED SYMBOLS DENOTE 1010 FT. DENS. ALT.
4. FLAGGED SYMBOLS DENOTE 975 FT. DENS. ALT.
5. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET.

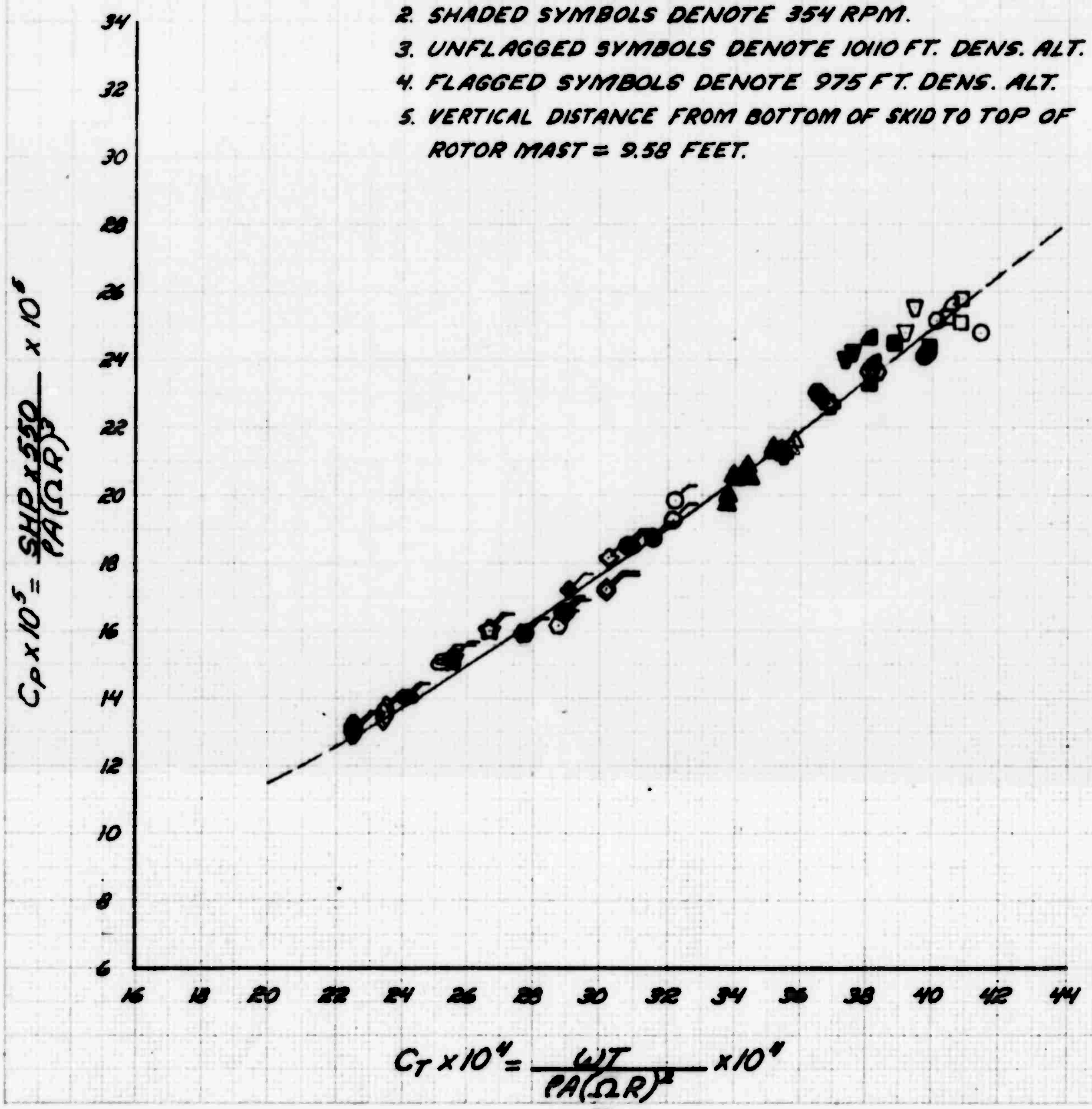


FIGURE 11
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811
SKID HEIGHT = 4 FEET
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS

SYM	GROSS WEIGHT (LB)	CONFIG	SYM	GROSS WEIGHT (LB)	CONFIG
○	3025	CLEAN	⬡	2535	ARMED
□	2790	ARMED	✪	2535	CLEAN
◇	2700	CLEAN	☆	2470	ARMED
◺	2640	ARMED	△	2330	CLEAN
▽	2610	CLEAN	⬭	2185	CLEAN

NOTE: 1. UNSHADED SYMBOLS DENOTE 347 RPM.
2. SHADED SYMBOLS DENOTE 354 RPM.
3. UNFLAGGED SYMBOLS DENOTE 10110 FT. DENS. ALT.
4. FLAGGED SYMBOLS DENOTE 1090 FT. DENS. ALT.
5. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET.

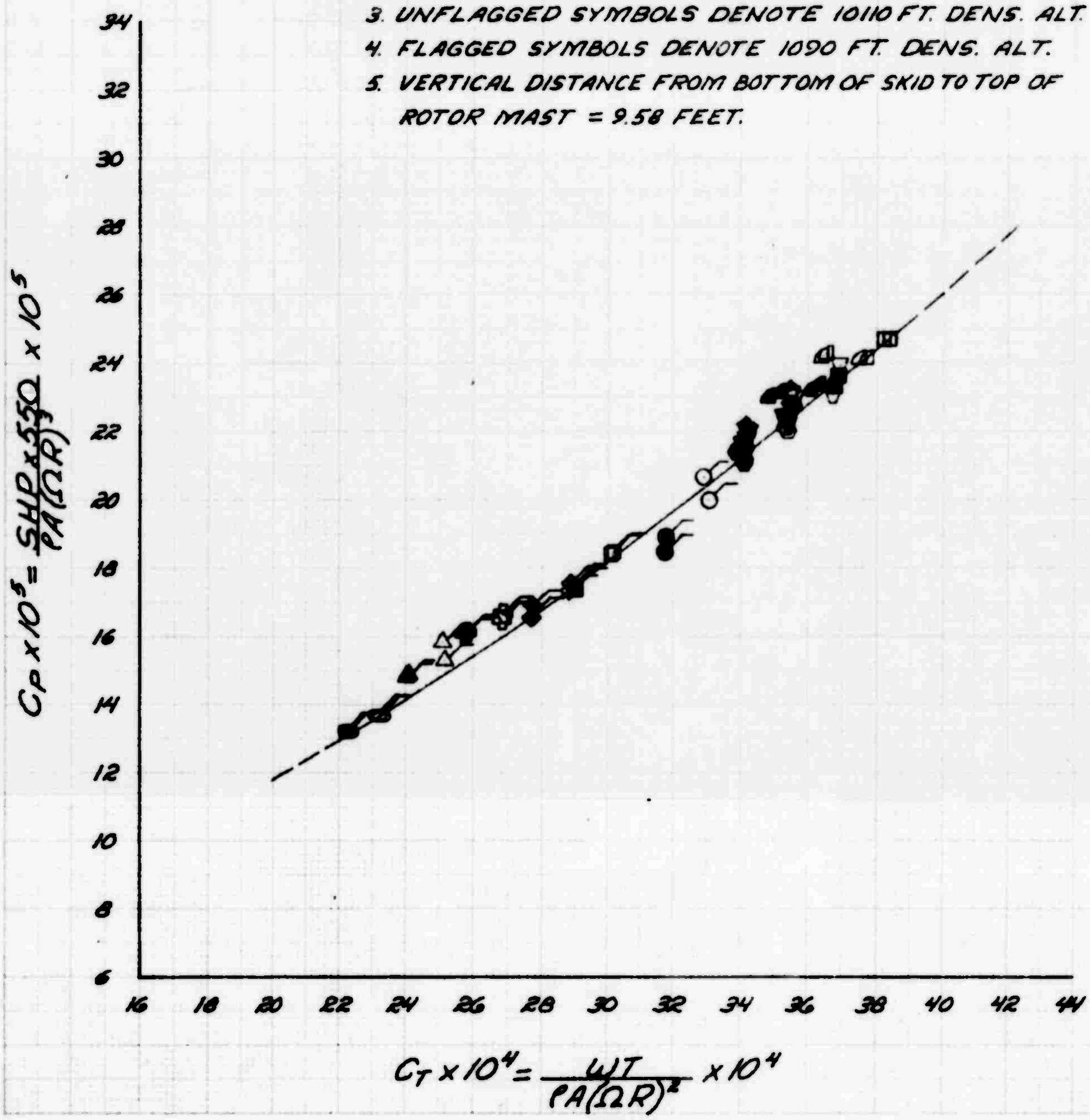


FIGURE 12
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811
SKID HEIGHT = 10 FEET
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS

SYM	GROSS WEIGHT (LB)	CONFIG	SYM	GROSS WEIGHT (LB)	CONFIG
○	3055	CLEAN	⬡	2320	ARMED
□	2965	ARMED	☆	2320	CLEAN
◇	2815	ARMED	△	2285	ARMED
◿	2710	CLEAN	⬭	2170	CLEAN
▽	2530	ARMED	⬯	2085	CLEAN
⬠	2530	CLEAN			

NOTE: 1. UNSHADED SYMBOLS DENOTE 347 RPM.
2. SHADED SYMBOLS DENOTE 354 RPM.
3. UNFLAGGED SYMBOLS DENOTE 9975 FT. DENS. ALT.
4. FLAGGED SYMBOLS DENOTE 4585 FT. DENS. ALT.
5. SLASHED SYMBOLS DENOTE 1075 FT. DENS. ALT.
6. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET.

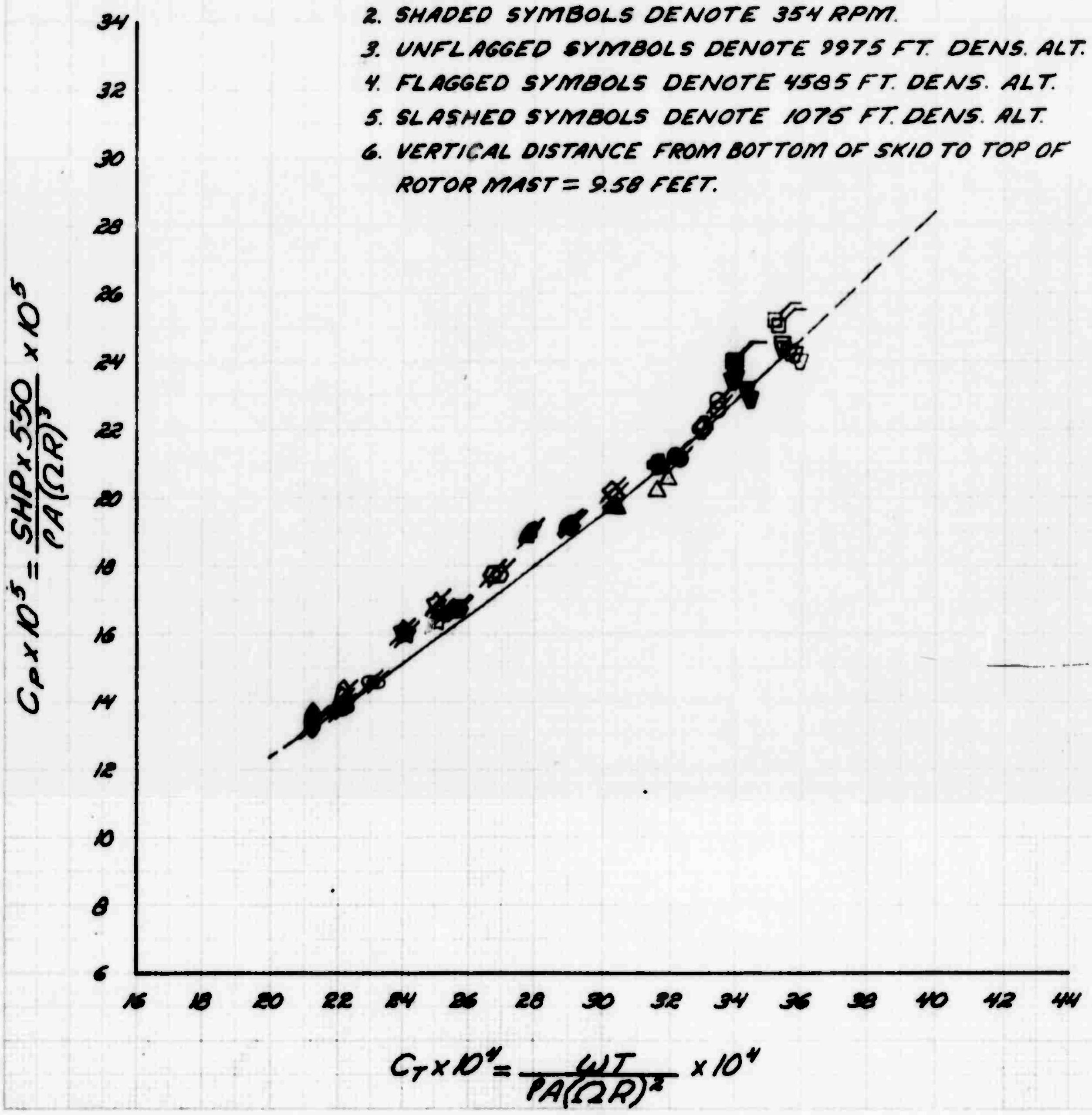


FIGURE 13
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811
SKID HEIGHT = 20 FEET
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS

SYM	GROSS WEIGHT (LB)	CONFIG	SYM	GROSS WEIGHT (LB)	CONFIG
○	3035	ARMED	○	2690	CLEAN
□	3035	CLEAN	⬡	2555	CLEAN
◇	2945	ARMED	⬠	2395	CLEAN
⏢	2875	ARMED	△	2285	ARMED
▽	2795	ARMED	⬭	2120	CLEAN

NOTE: 1. UNSHADED SYMBOLS DENOTE 347 RPM.
2. SHADED SYMBOLS DENOTE 354 RPM.
3. UNFLAGGED SYMBOLS DENOTE 9430 FT. DENS. ALT.
4. FLAGGED SYMBOLS DENOTE 4295 FT. DENS. ALT.
5. SLASHED SYMBOLS DENOTE 830 FT. DENS. ALT.
6. V. RTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET.

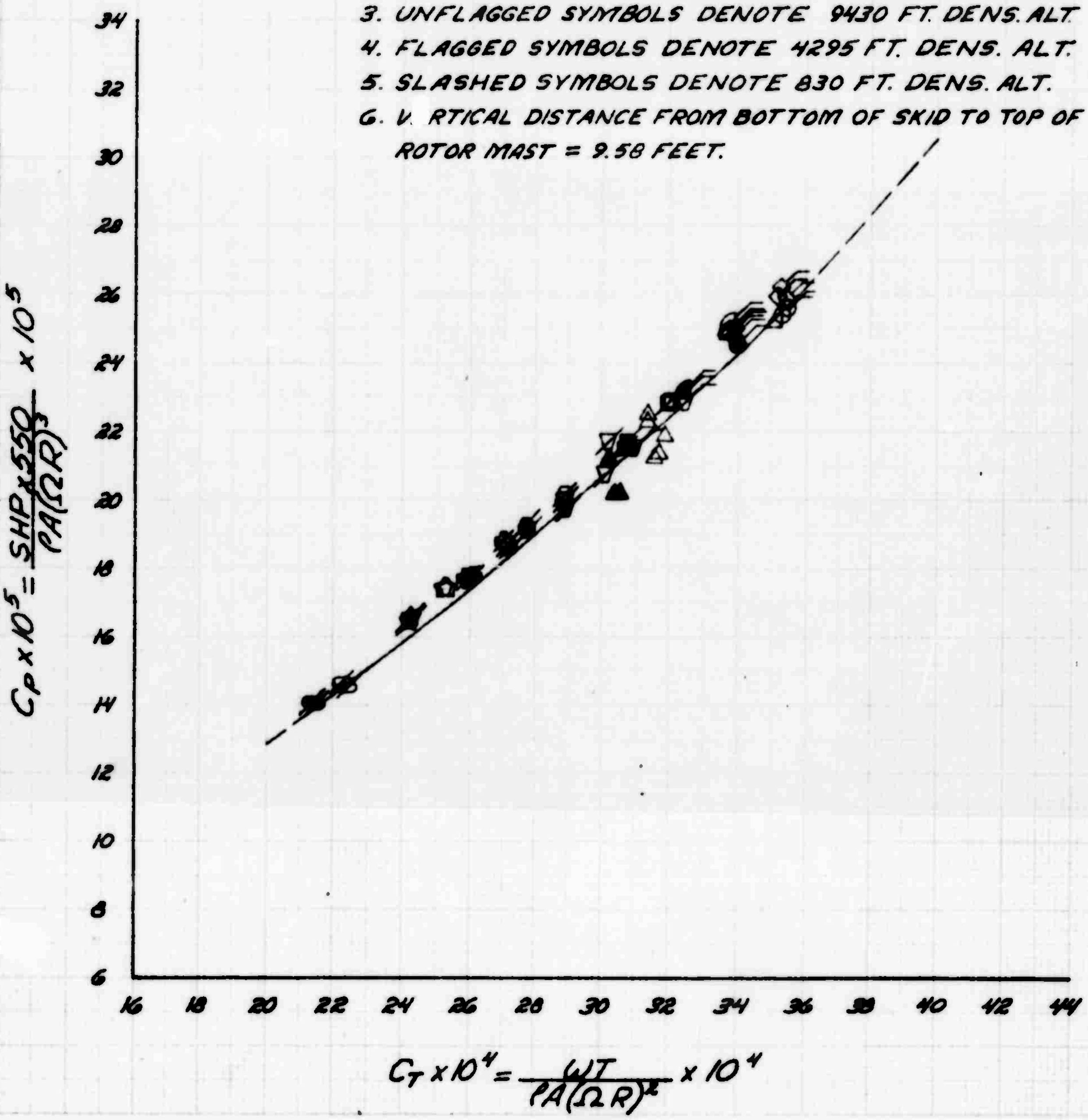


FIGURE 14
NON-DIMENSIONAL HOVERING PERFORMANCE
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811
SKID HEIGHT = 50 FEET
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS

SYM	GROSS WEIGHT (LB)	CONFIG	SYM	GROSS WEIGHT (LB)	CONFIG
○	3050	CLEAN	✧	2440	CLEAN
□	2955	ARMED	☆	2405	CLEAN
◇	2690	ARMED	△	2305	ARMED
◺	2665	CLEAN	⬭	2305	CLEAN
▽	2585	CLEAN	0	2100	CLEAN
○	2545	CLEAN			

NOTE: 1. UNSHADED SYMBOLS DENOTE 347 RPM.
2. SHADED SYMBOLS DENOTE 354 RPM.
3. UNFLAGGED SYMBOLS DENOTE 9385 FT. DENS. ALT.
4. FLAGGED SYMBOLS DENOTE 3340 FT. DENS. ALT.
5. SLASHED SYMBOLS DENOTE 235 FT. DENS. ALT.
6. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET.

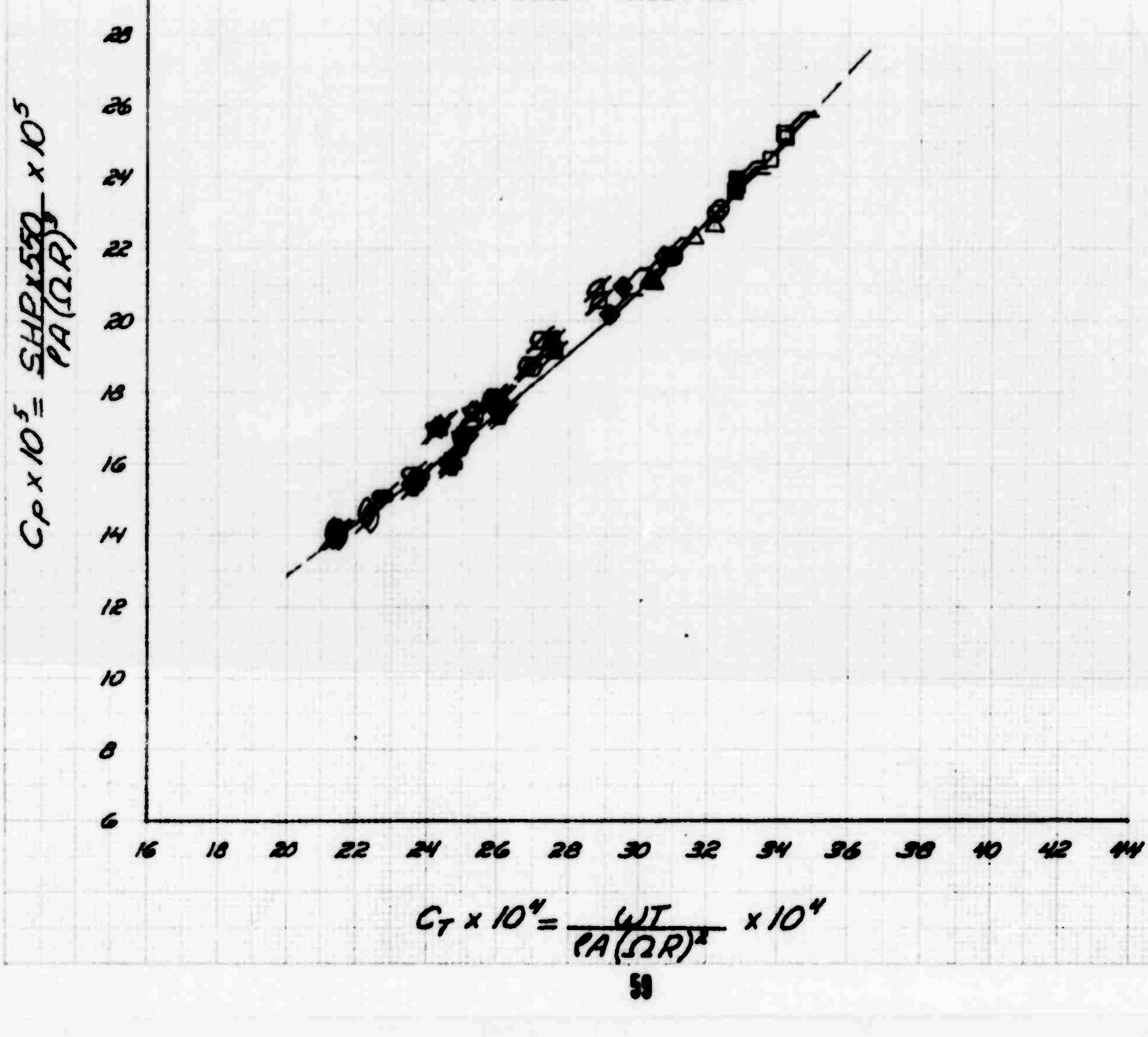


FIGURE 15
CLIMB PERFORMANCE
OH-58A USA S/N68-16693
T63-A-700 ENGINE
TAKEOFF RATED POWER T_{T_5} = 1380 °F
STANDARD DAY

TAKEOFF GROSS WEIGHT = 2747 LB CG = 107.0 IN. (FWD) ROTOR SPEED = 354 RPM CLEAN CONFIGURATION

NOTE: 1. TEST POINTS DENOTE TWO SEPARATE CLIMBS.

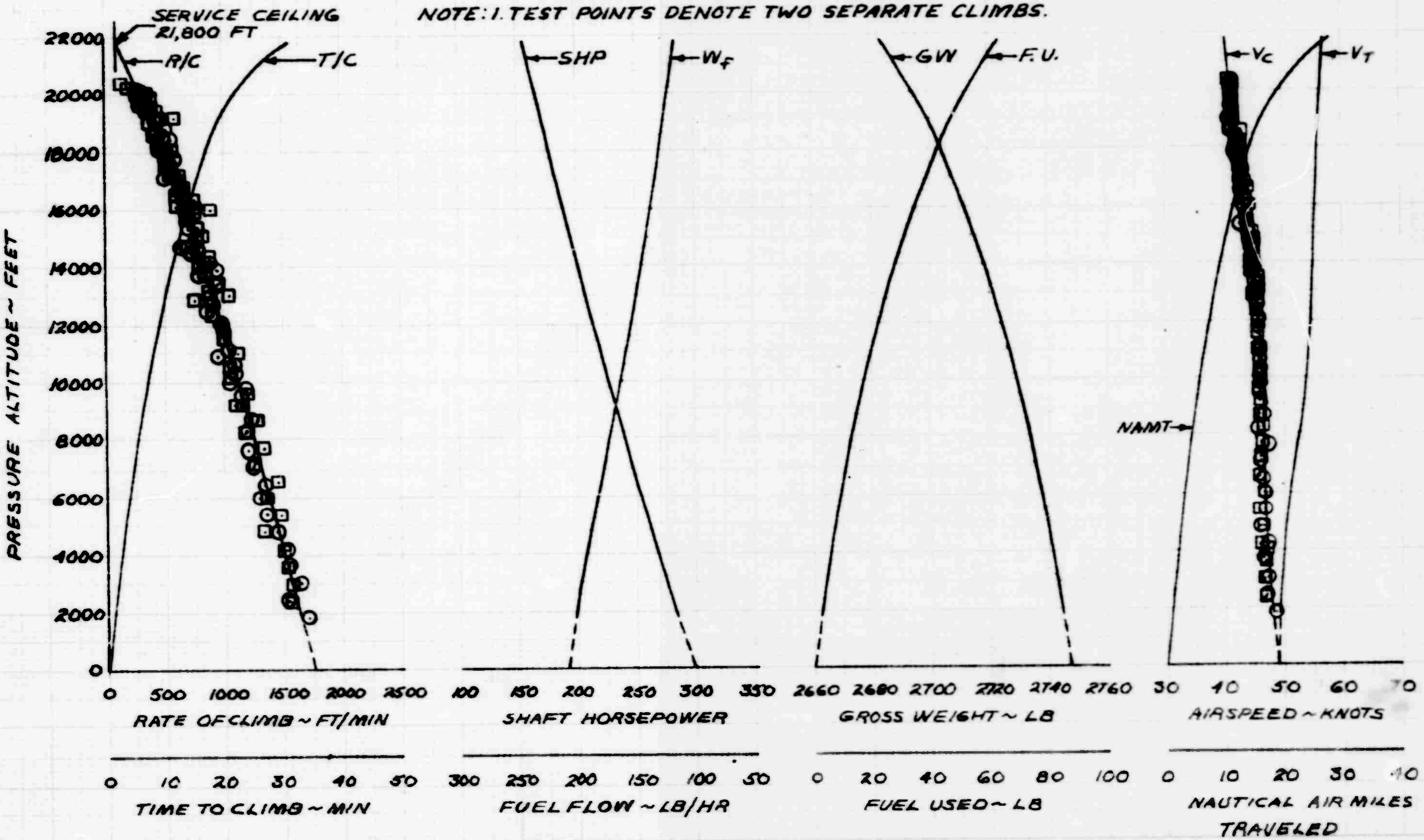


FIGURE 16A
CLIMB PERFORMANCE
OH-58A U.S.A. S/N 68-16693
T63-A-700 ENGINE
TAKEOFF RATED POWER T_{T5} = 1380°F
STANDARD DAY

TAKEOFF GROSS WEIGHT = 2252 LBS   ROTOR SPEED = 354 RPM
C.G. STATION = 107.0 IN. (FWD)   CLEAN CONFIGURATION

NOTE: 1. TEST POINTS DENOTE TWO SEPARATE CLIMBS.

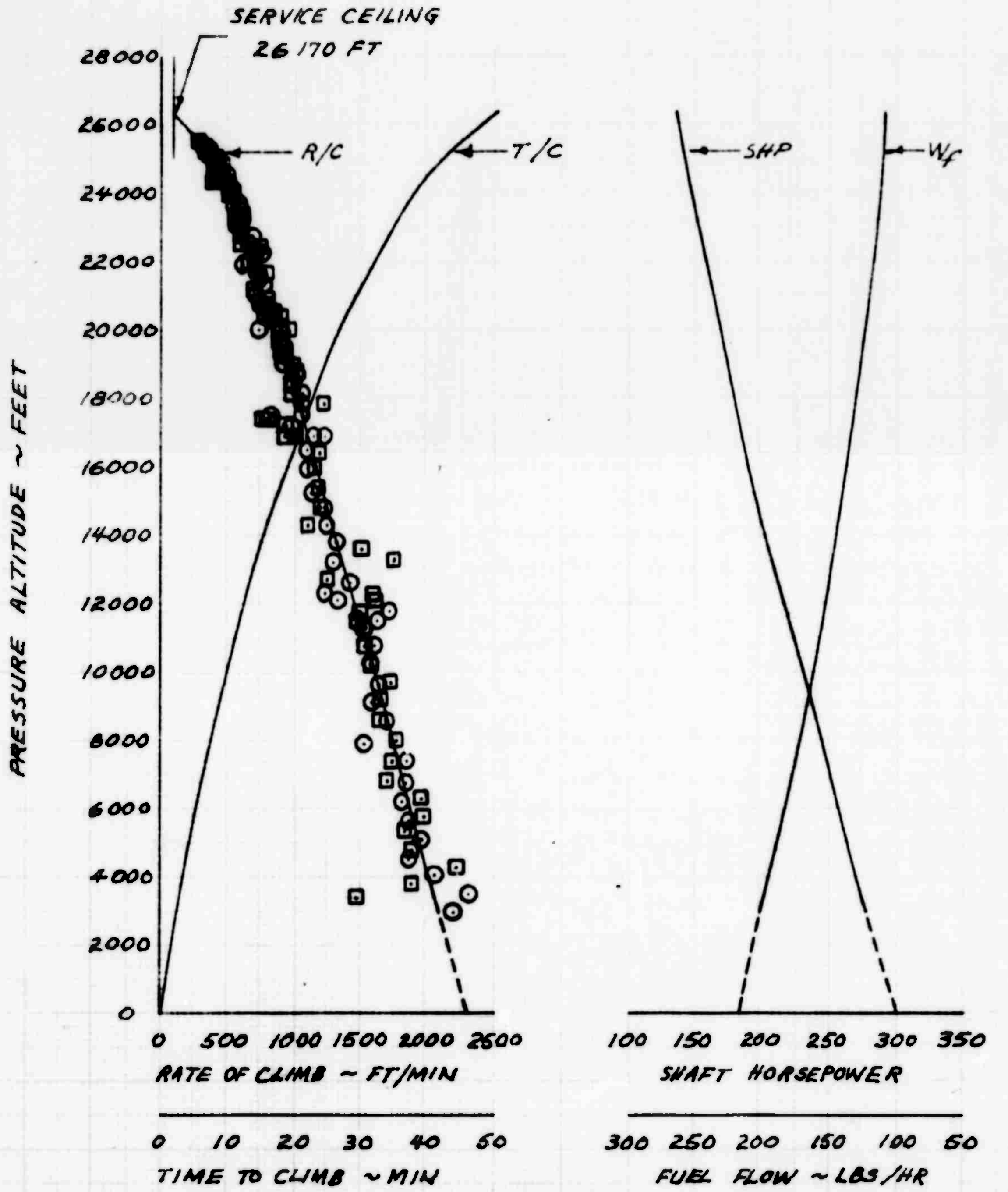


FIGURE 16B

CLIMB PERFORMANCE (CONCLUDED)

OH-58A U.S.A. S/N 68-16693

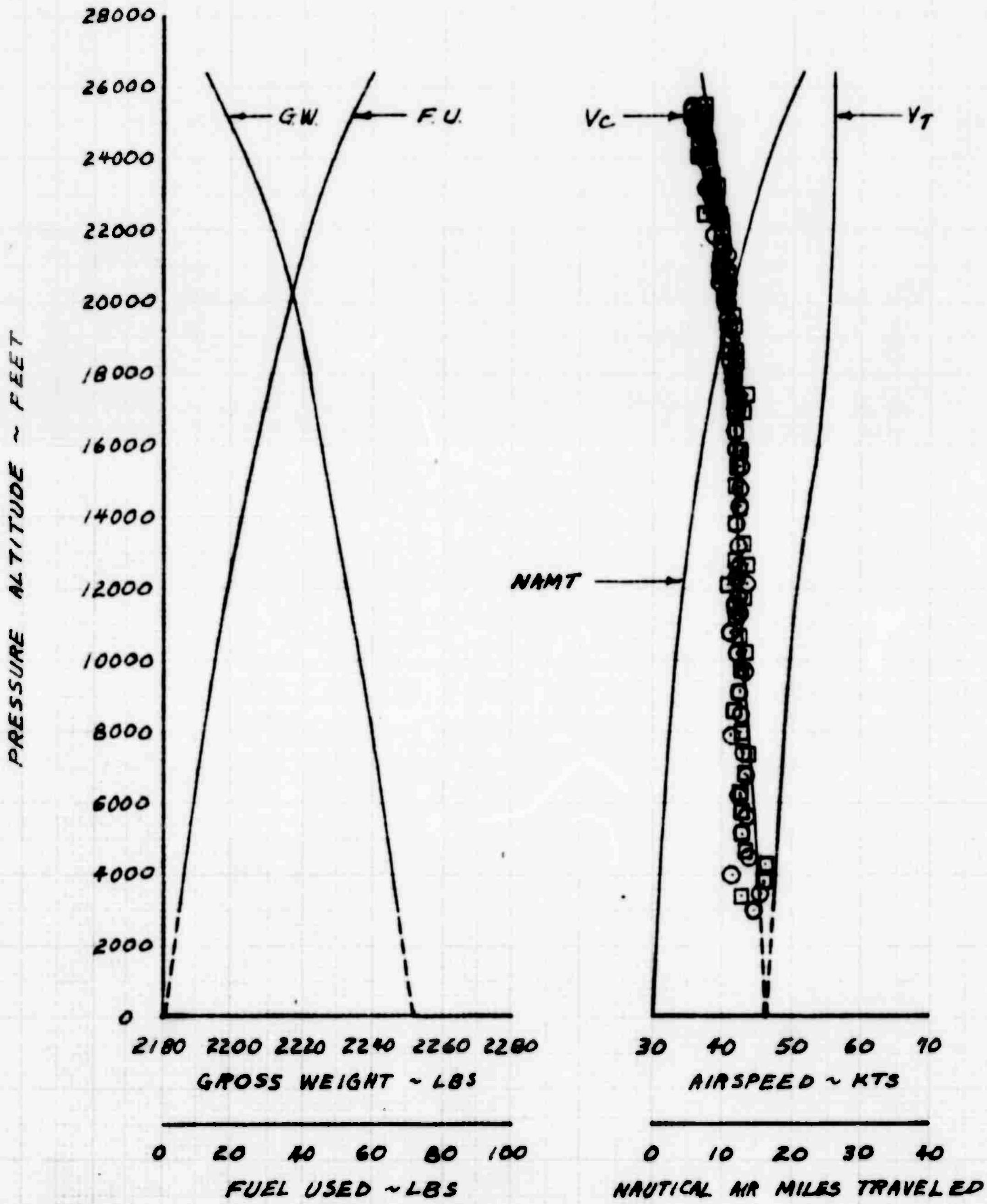


FIGURE 17
CLIMB PERFORMANCE
OH-58A U.S.A. S/N 68-16693
T63-A-700 ENGINE
TAKEOFF RATED POWER T_{T_5} = 1380 °F
STANDARD DAY

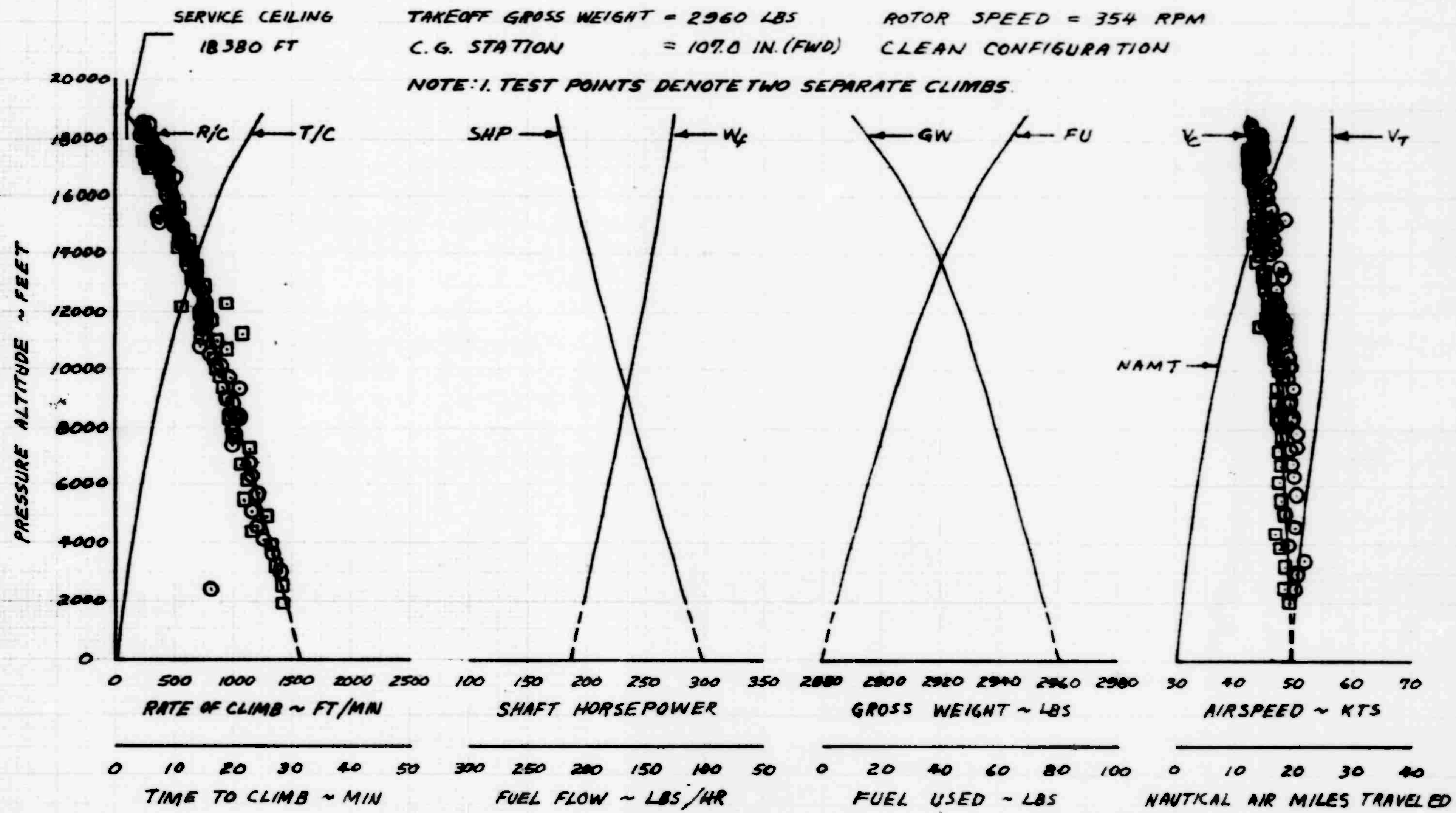


FIGURE 18
CLIMB PERFORMANCE
OH-58A USA S/N68-16693
T63-A-700 ENGINE
NORMAL RATED POWER T_{T5} = 1280°F
STANDARD DAY

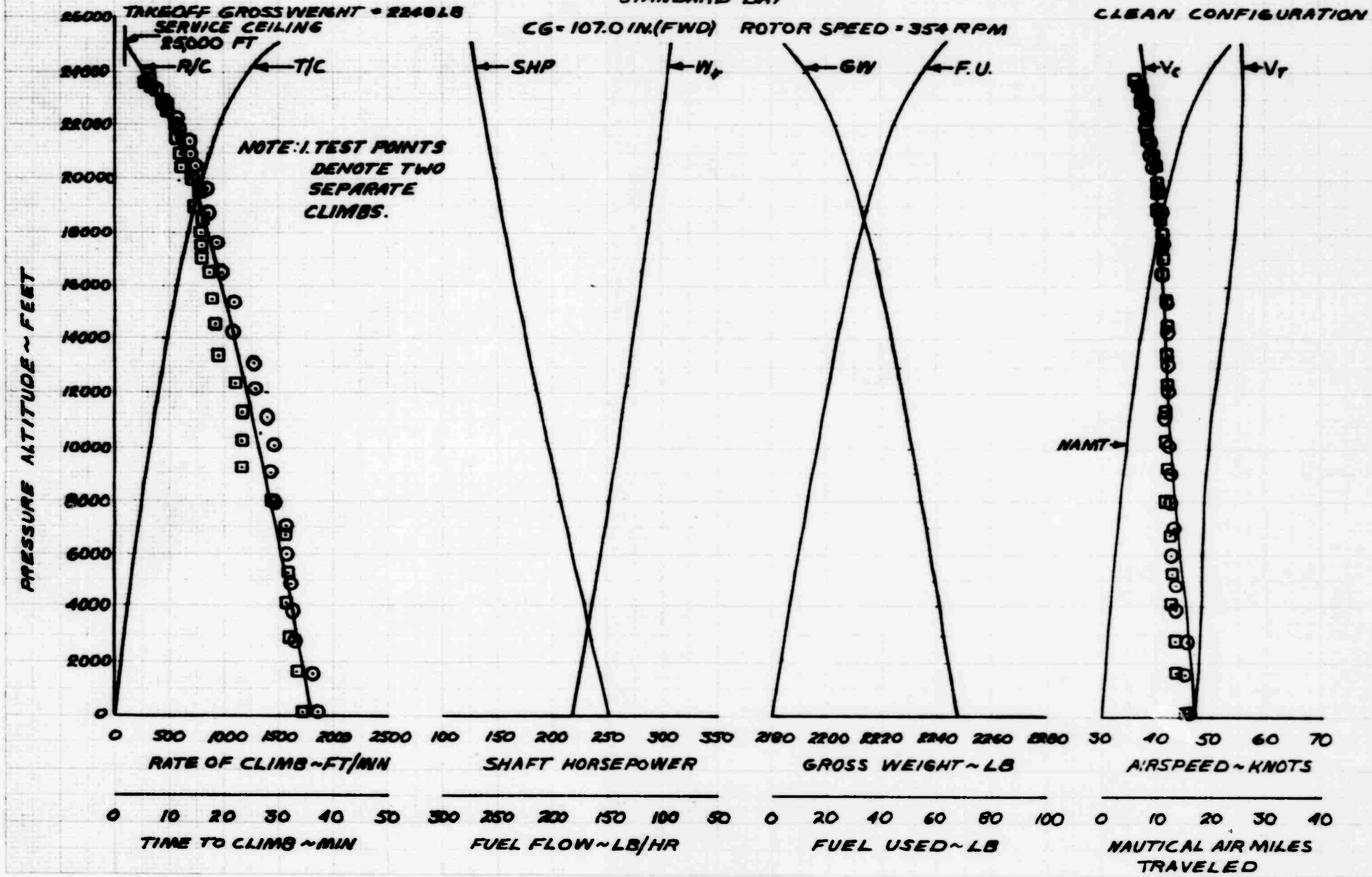


FIGURE 19
CLIMB PERFORMANCE
OH-58A USA S/N68-16693
T63-A-700 ENGINE
NORMAL RATED POWER T_{T_5} = 1280°F
STANDARD DAY

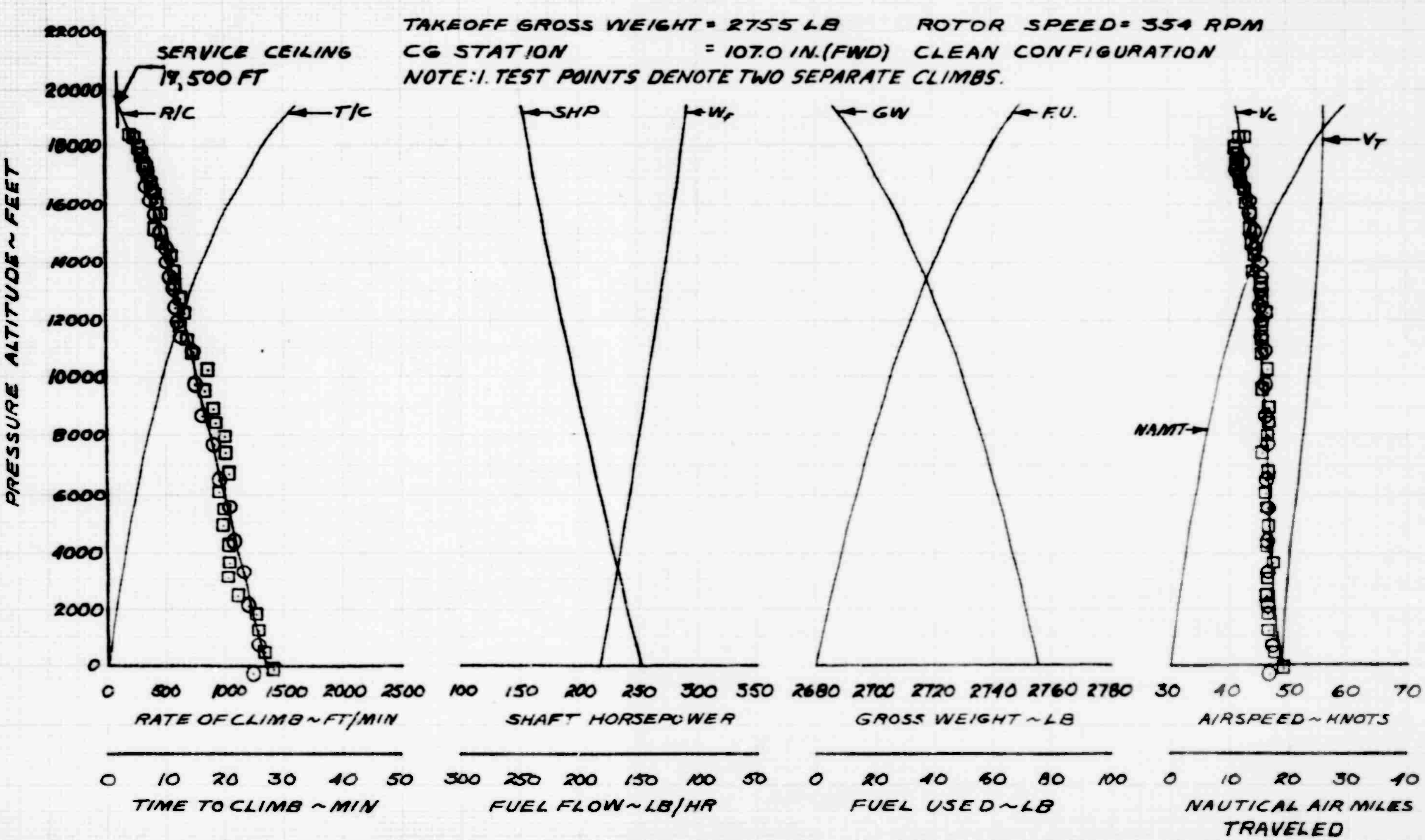


Figure 20

Climb Performance

OH-58A USAS/N68-16693

T63-A-700 ENGINE

NORMAL RATED POWER TT_6 = 1280°F

STANDARD DAY

TAKEOFF GROSS WEIGHT = 2965 LB ROTOR SPEED = 354 RPM

CG STATION = 107.0 IN.(FWD) CLEAN CONFIGURATION

NOTE: 1. TEST POINTS DENOTE TWO SEPARATE CLIMBS.

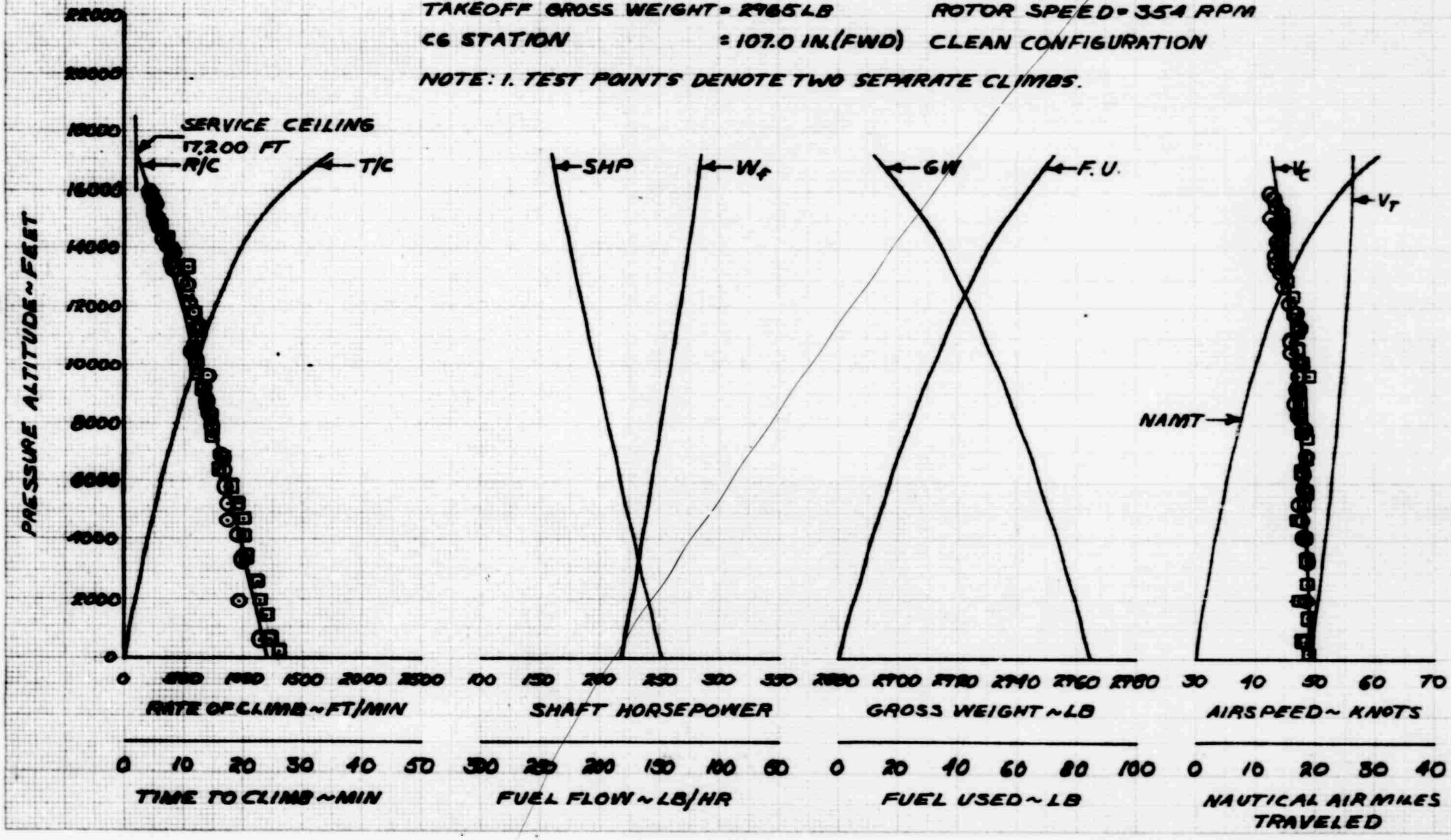


FIGURE 21
VARIATION IN RATE OF CLIMB
AS A FUNCTION OF SHAFT HORSEPOWER
OH-58A USA S/N 68-16693

SYM	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)
○	4670	2560
□	9865	2560

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

$$K_p = 0.7946$$

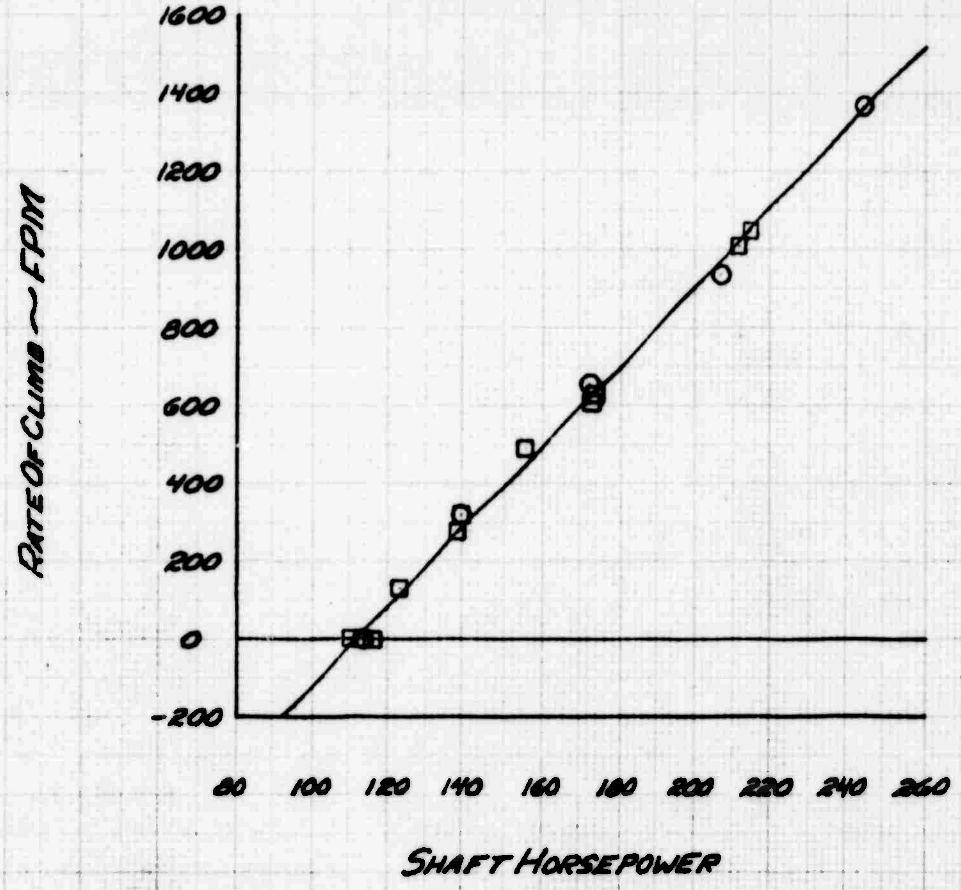
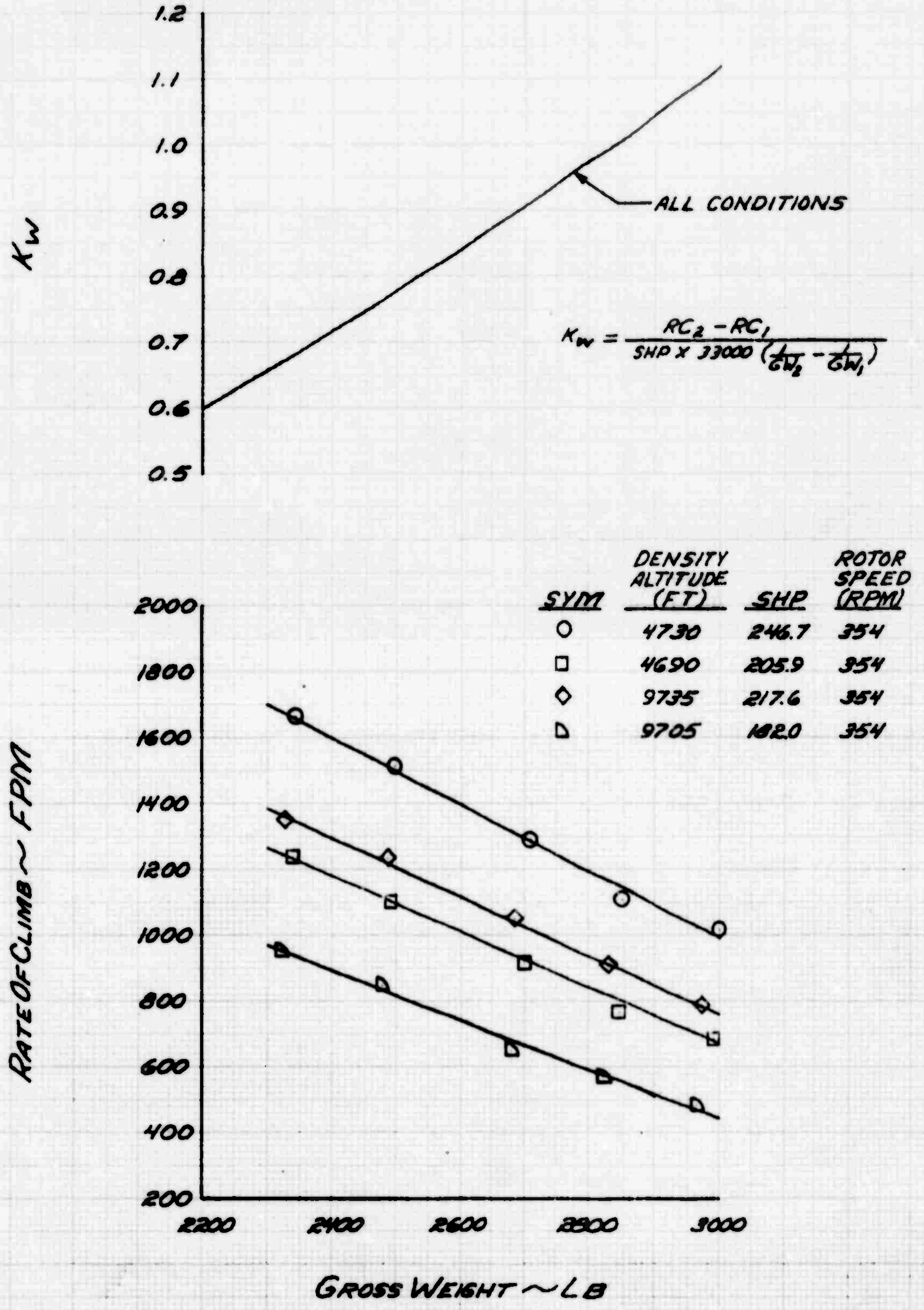


FIGURE 22
VARIATION IN RATE OF CLIMB
AS A FUNCTION OF GROSS WEIGHT
OH-58A USA S/N 68-16698



SYM	DENSITY ALTITUDE (FT)	SHP	ROTOR SPEED (RPM)
○	4730	246.7	354
□	4690	205.9	354
◇	9735	217.6	354
◺	9705	182.0	354

FIGURE 23
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693
CLEAN-ALL DOORS ON
CG = 107.0 IN. (FWD)

NOTE: 1. CURVES DERIVED FROM FIGS. 32 THRU 44
2. SHADED SYMBOLS DENOTE COMPRESSIBILITY EFFECTS.

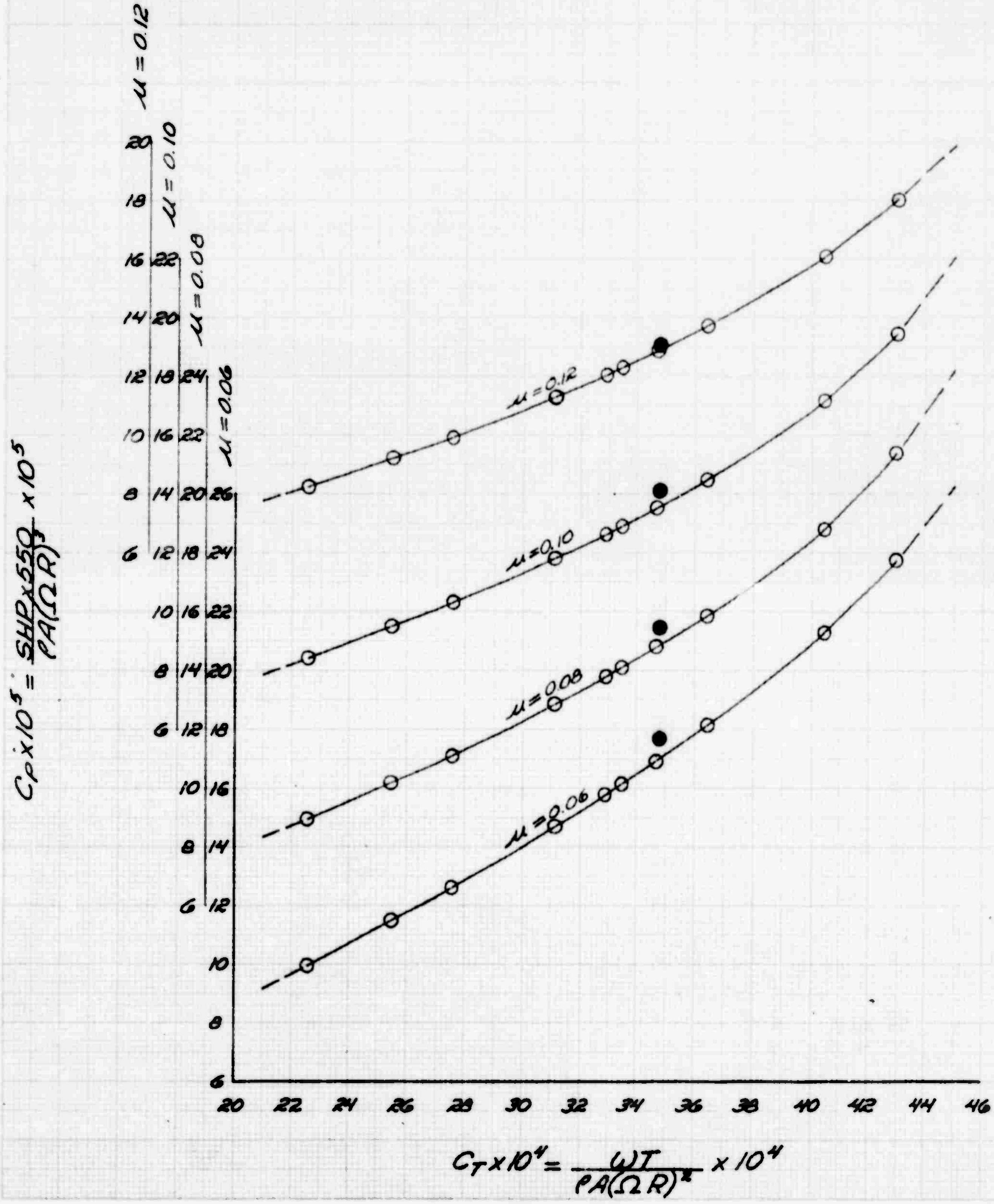
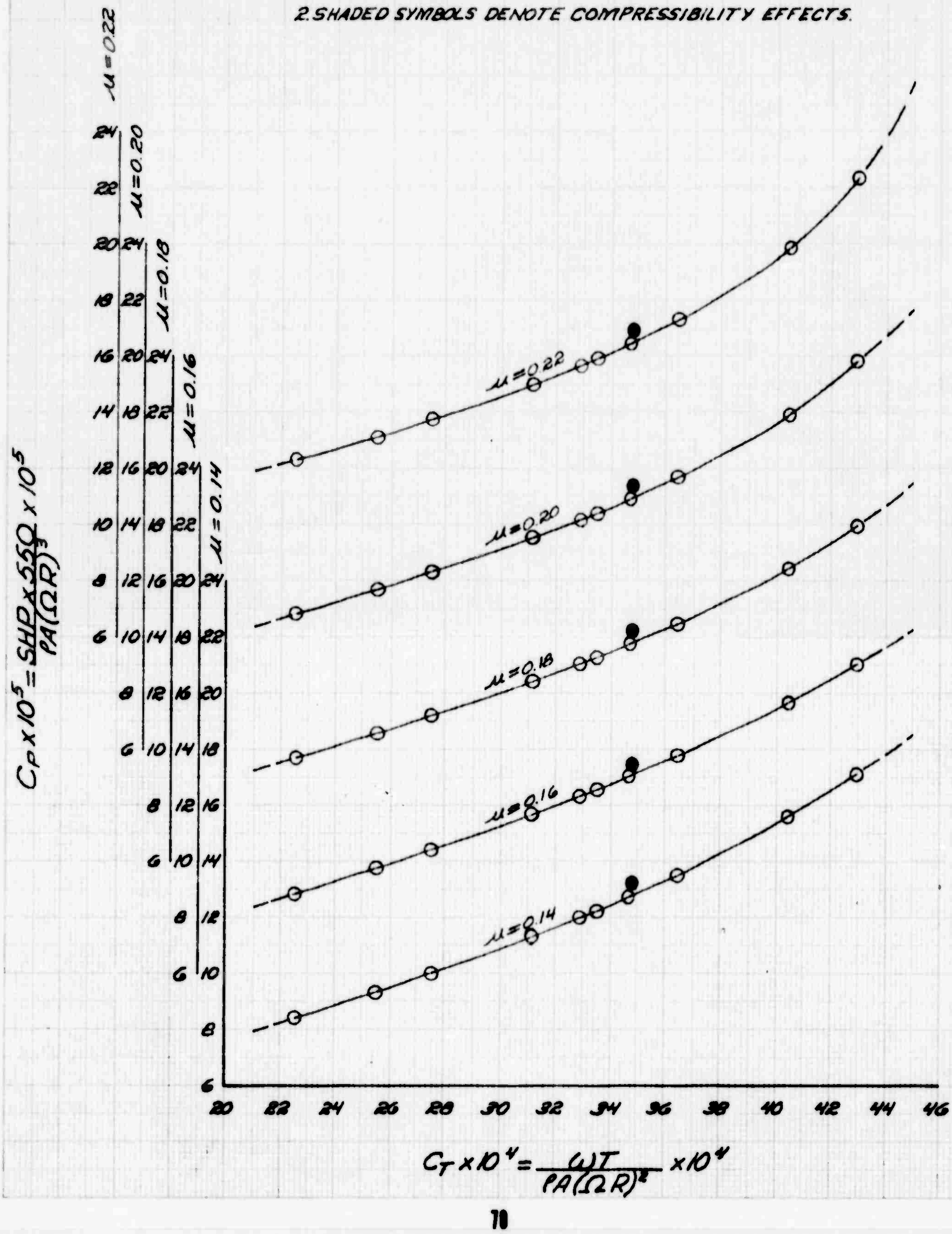


FIGURE 24
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693
CLEAN-ALL DOORS ON
CG = 107.0 IN. (FWD)

NOTE: 1. CURVES DERIVED FROM FIGS. 32 THRU 44
2. SHADED SYMBOLS DENOTE COMPRESSIBILITY EFFECTS.



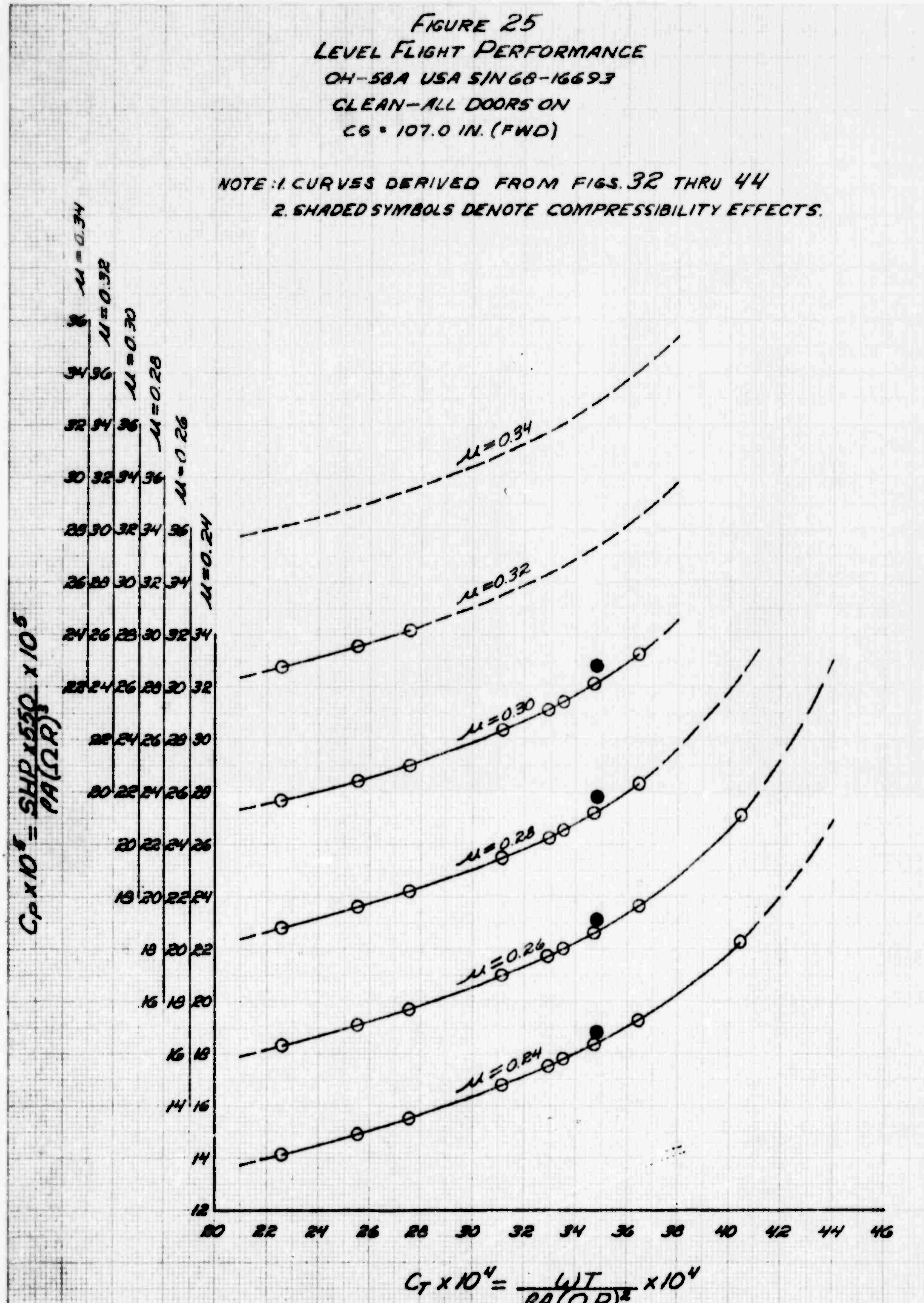


FIGURE 25
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693
CLEAN-ALL DOORS ON
CG = 107.0 IN. (FWD)
NOTE: 1. CURVES DERIVED FROM FIGS. 32 THRU 44
2. SHADED SYMBOLS DENOTE COMPRESSIBILITY EFFECTS.
μ = 0.34
μ = 0.32
μ = 0.30
μ = 0.28
μ = 0.26
μ = 0.24
$C_P \times 10^5 = \frac{SHP \times 550}{\rho A(\Omega R)^3} \times 10^5$
$C_T \times 10^4 = \frac{W T}{\rho A(\Omega R)^2} \times 10^4$

FIGURE 26
LEVEL FLIGHT RANGE SUMMARY
OH-58A USA S/N 68-16693
SEA LEVEL — STANDARD DAY
CG = 107.0 IN. (FWD)
NOTE: 1. DATA DERIVED FROM FIGURES 23 THRU 25 AND 72

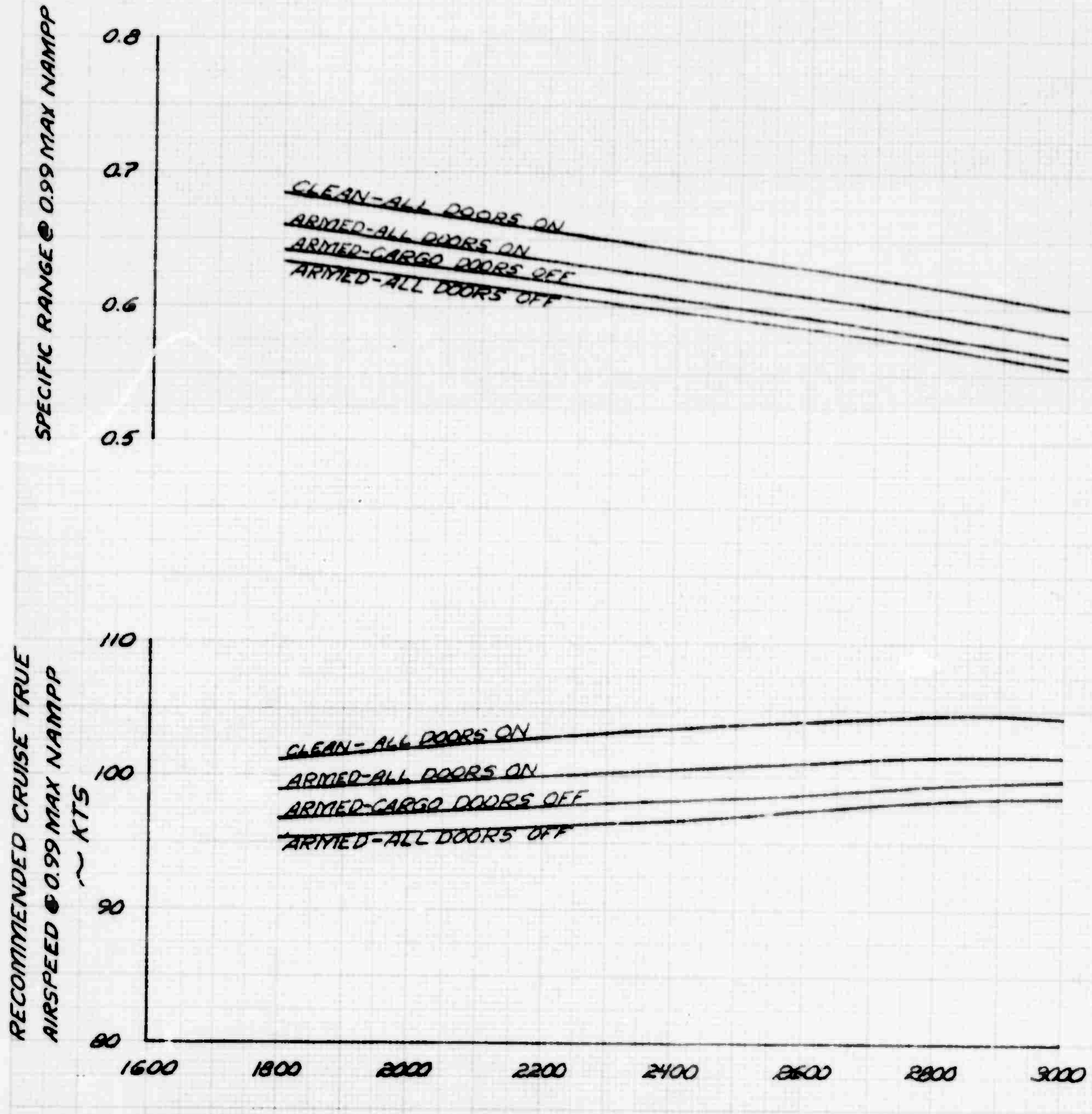


FIGURE 27
LEVEL FLIGHT RANGE SUMMARY
OH-58A USA S/N 68-16693
5000 FEET STANDARD DAY
CG = 107.0 IN. (FWD)
NOTE: 1. DATA DERIVED FROM FIGURES 23 THRU 25 AND 72
2. DASHED LINES DENOTE NORMAL RATED POWER LIMIT.

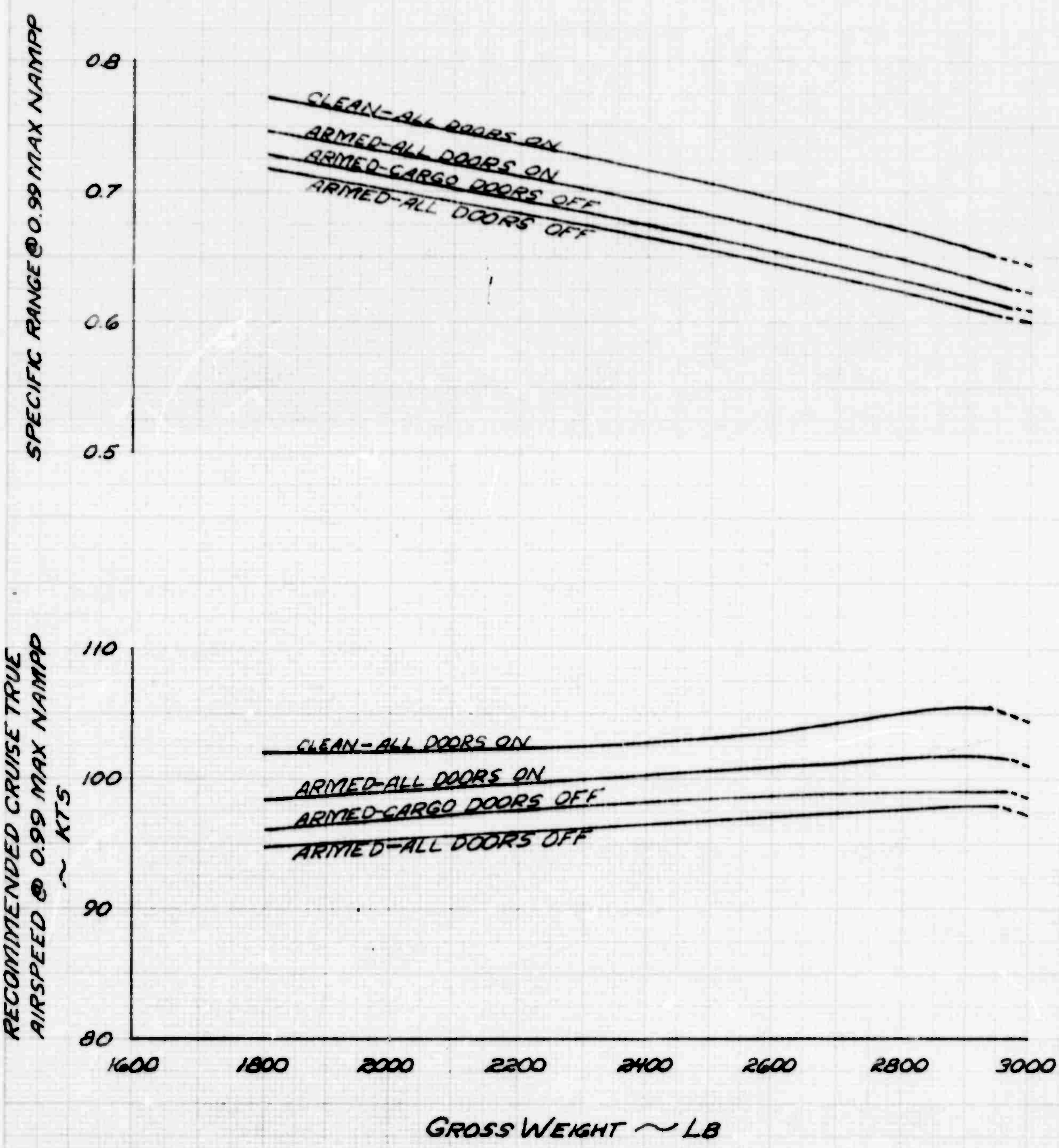


FIGURE 28
LEVEL FLIGHT RANGE SUMMARY
OH-58A USA S/N 68-16693
10000 FEET STANDARD DAY
CG = 107.0 IN. (FWD)

NOTE: 1. DATA DERIVED FROM FIGURES 23 THRU 25 AND 72
2. DASHED LINE DENOTES NORMAL RATED POWER LIMIT.

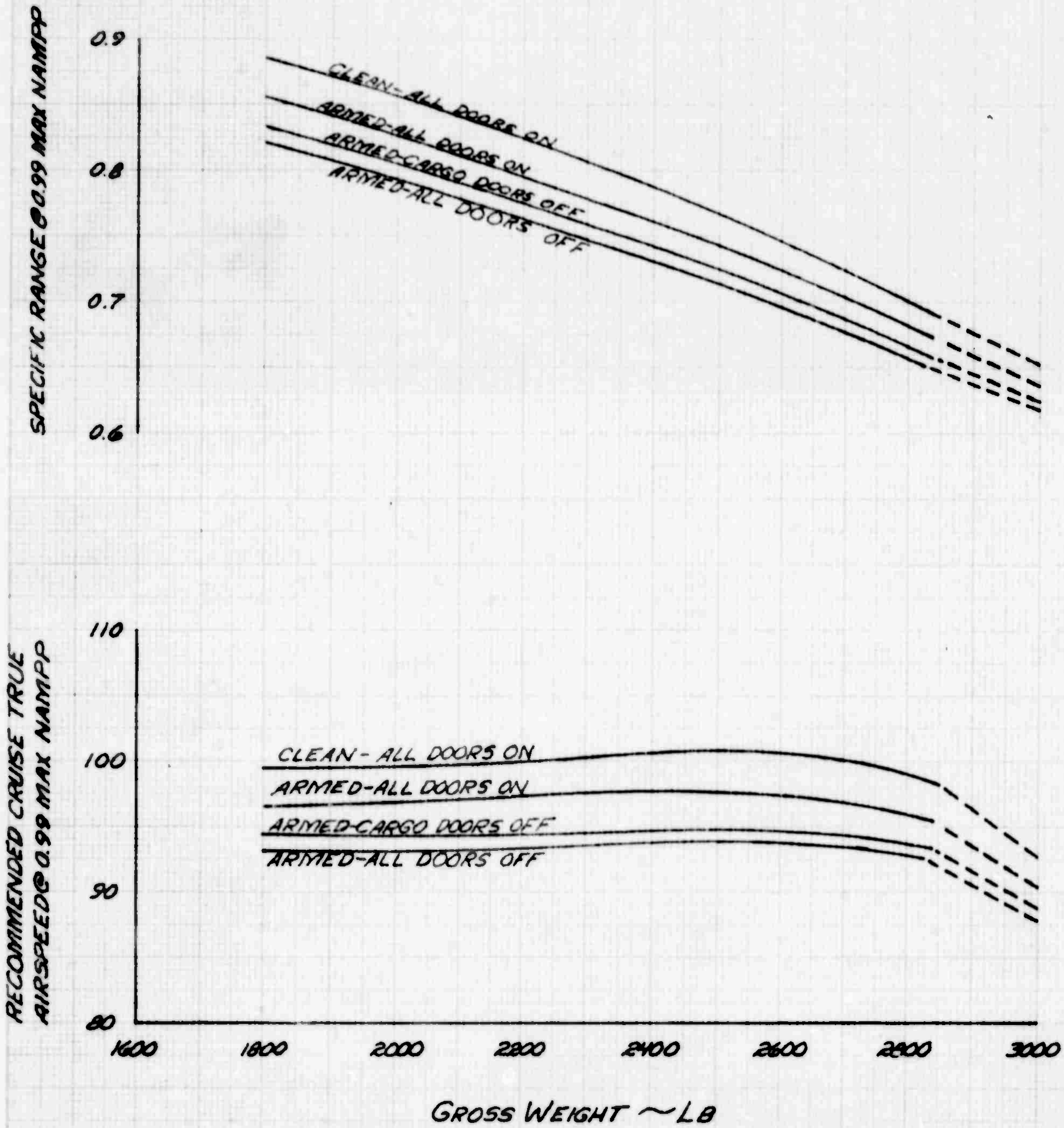
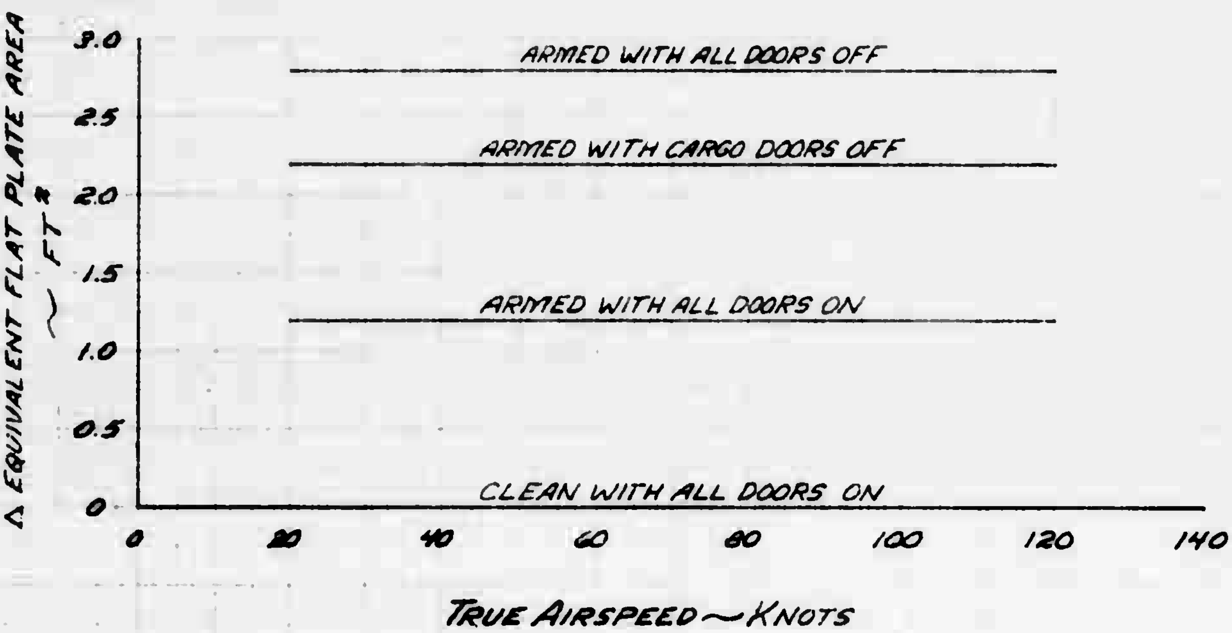


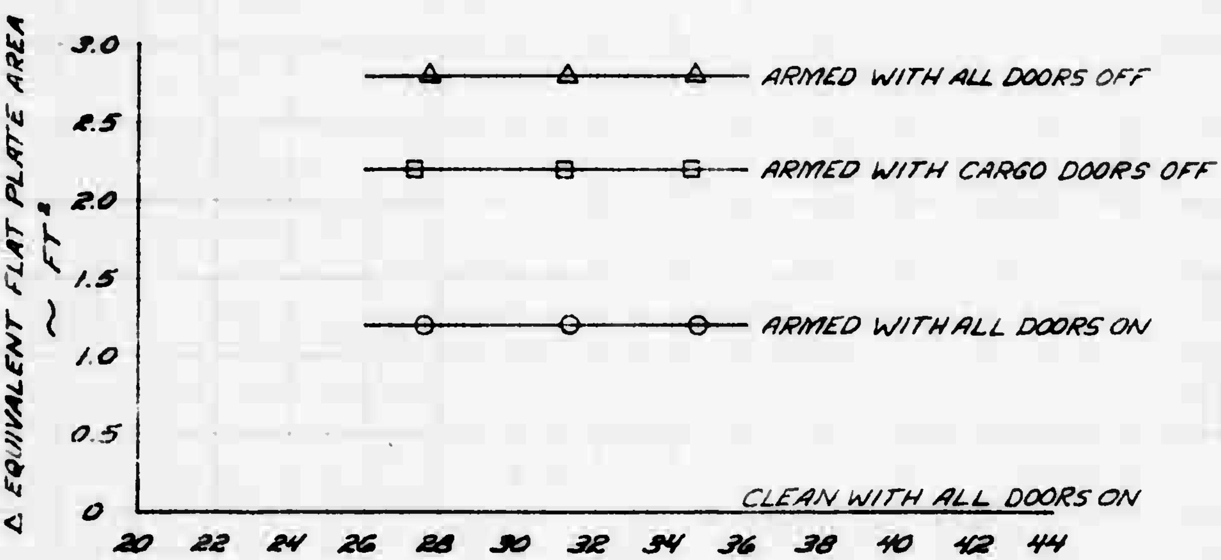
FIGURE 29

LEVEL FLIGHT PERFORMANCE WITH CONFIGURATION CHANGE

OH-58A USA S/N 68-16693

NOTE: 1. DATA OBTAINED FROM FIGURES 45 THRU 53

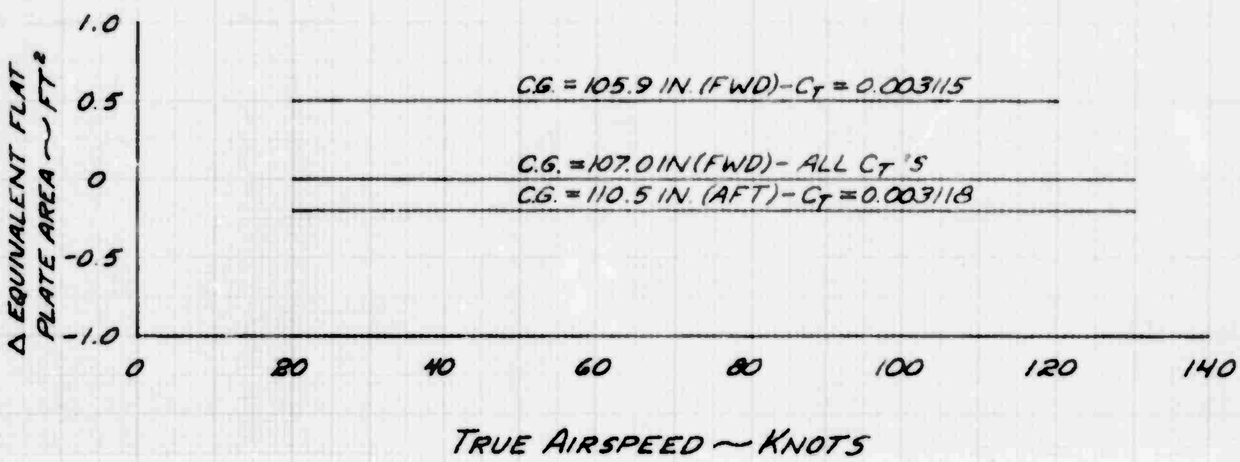




$$C_T \times 10^4 = \frac{W T}{\rho A (\Omega R)^2} \times 10^4$$

FIGURE 30
LEVEL FLIGHT PERFORMANCE WITH C.G. CHANGE
OH-58A USA S/N 68-16693
CLEAN-ALL DOORS ON

NOTE: 1. DATA OBTAINED FROM FIGURES 36 AND 37



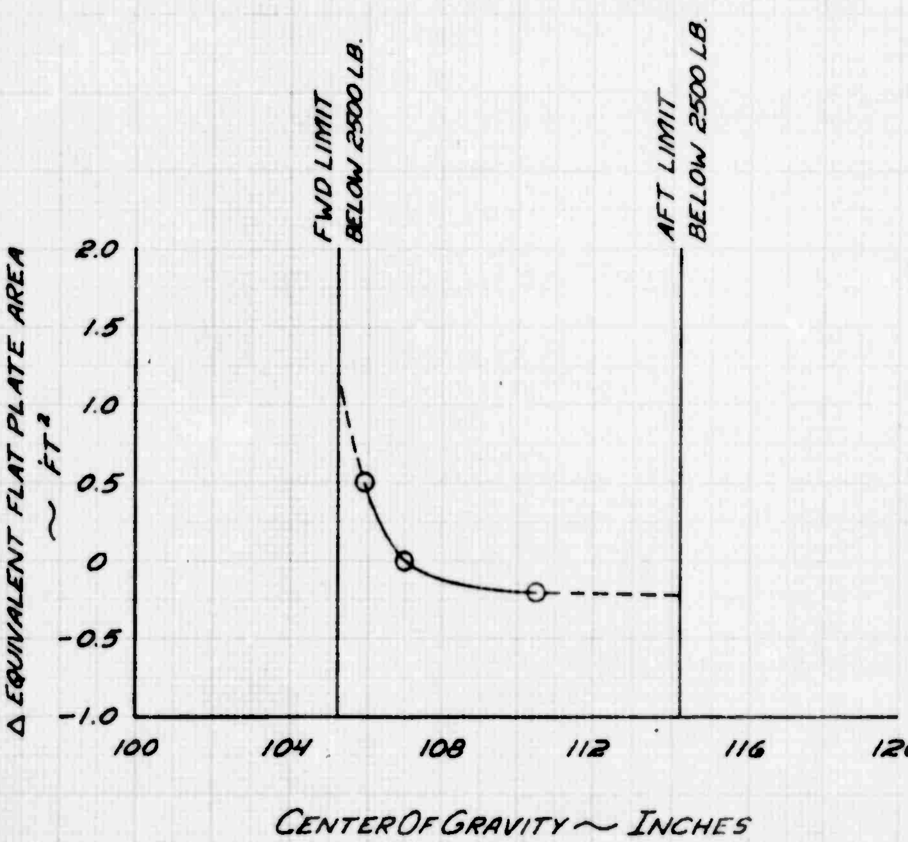


FIGURE 31

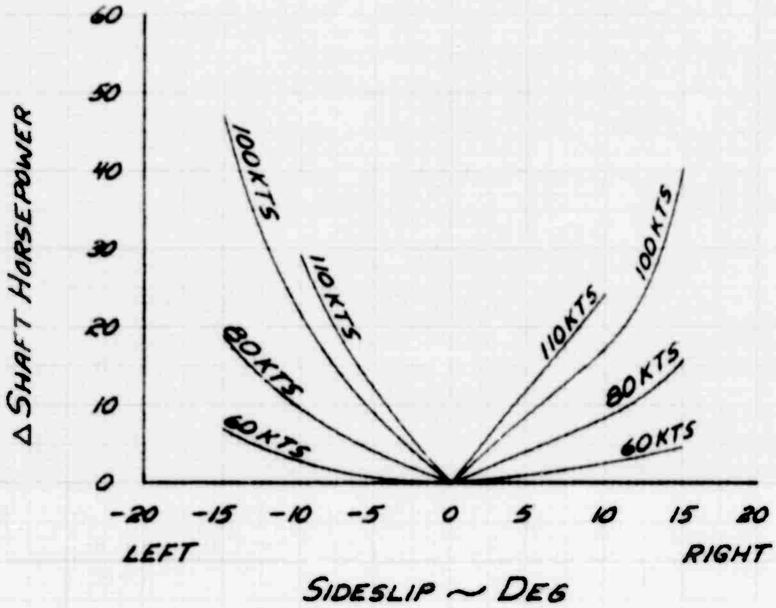
SIDESLIP EFFECTS ON LEVEL FLIGHT PERFORMANCE

OH-58A USA S/N 68-16693

CLEAN-ALL DOORS ON

DENSITY ALTITUDE = 4780 FT.
ROTOR SPEED = 354 RPM
GROSS WEIGHT = 2682 LB.
C.G. STATION = 106.9 IN. (FWD)

$C_T = 0.003090$



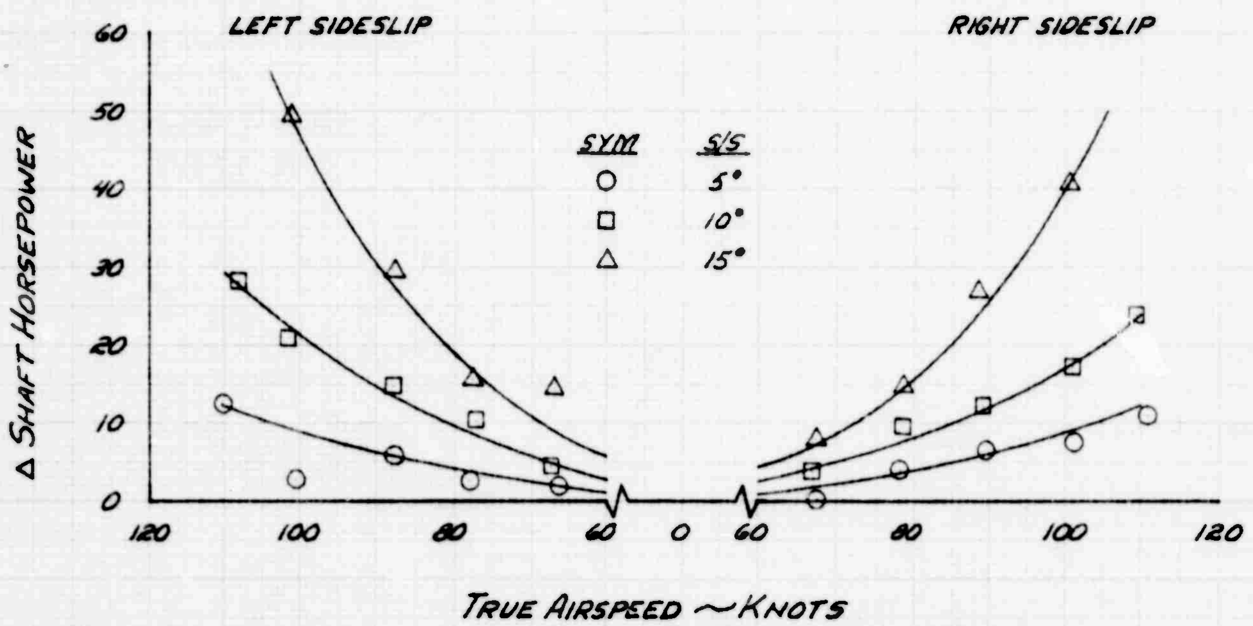
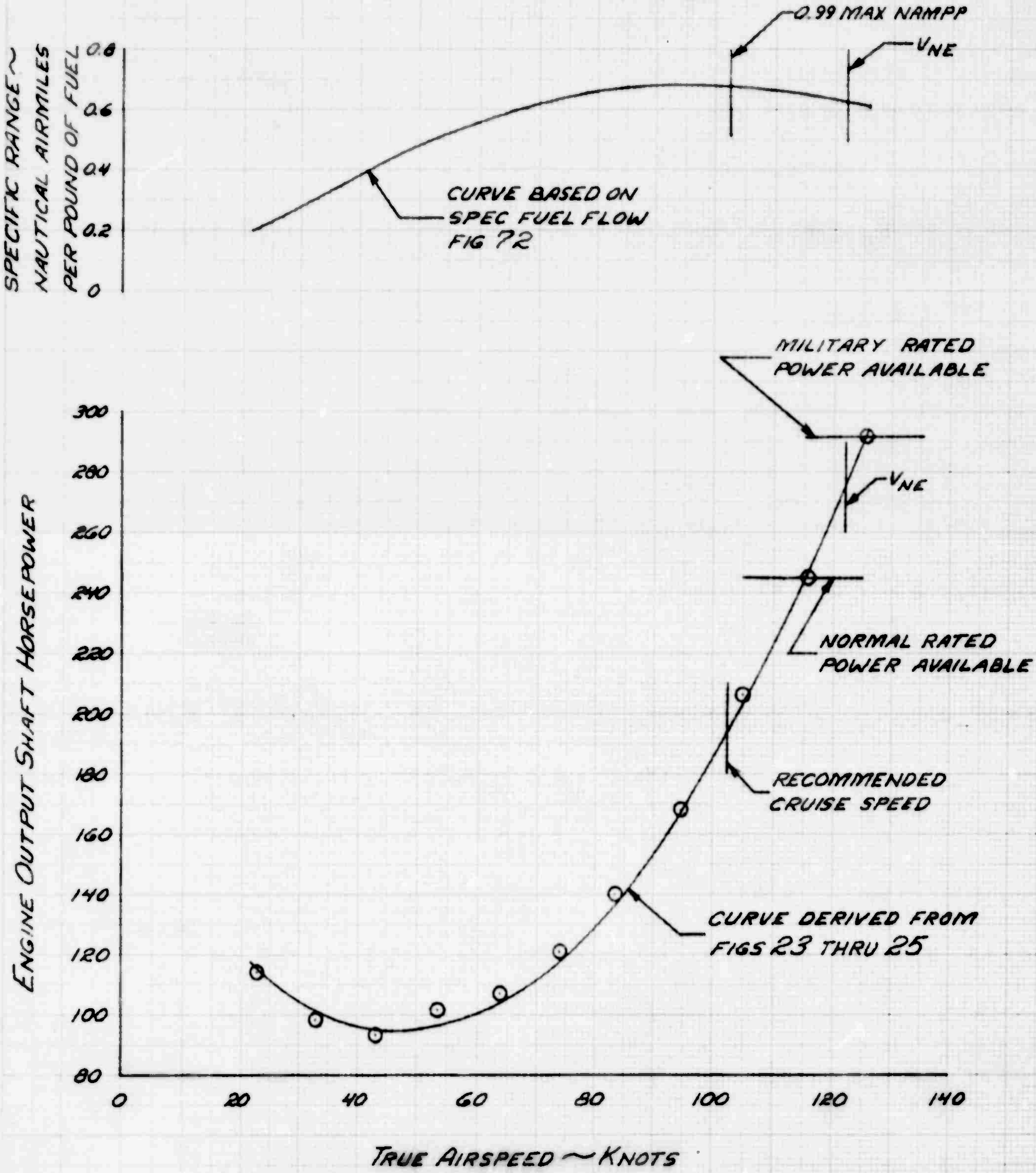


FIGURE 32
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 1150 FT	CG STATION	= 107.0 IN (FWD)
GROSS WEIGHT	= 2180 LB	C_T	= 0.002254
ROTOR SPEED	= 354 RPM	CONFIGURATION	= CLEAN - ALL DOORS ON



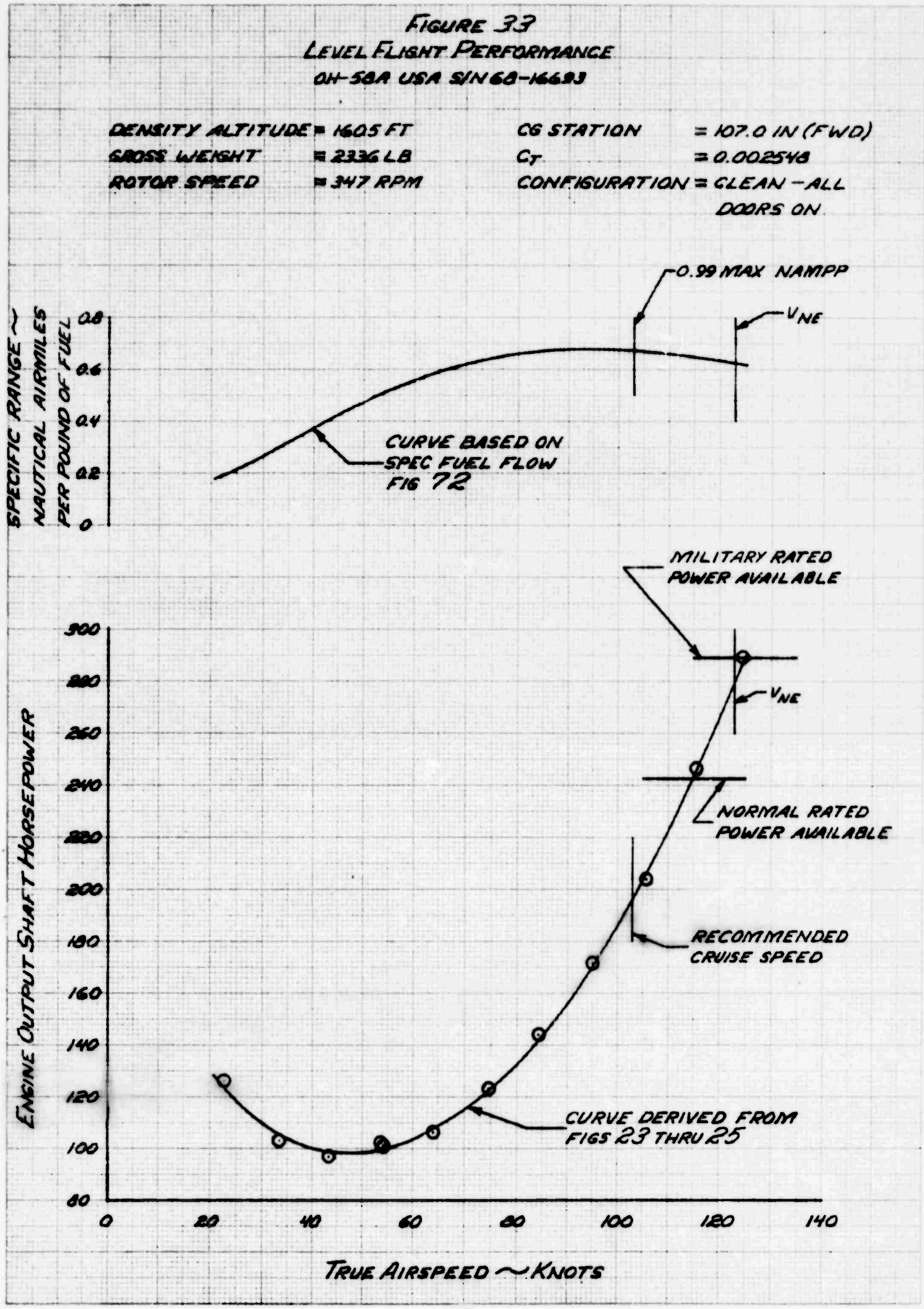


FIGURE 33
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693
DENSITY ALTITUDE = 1605 FT
GROSS WEIGHT = 2336 LB
ROTOR SPEED = 347 RPM
CG STATION = 107.0 IN (FWD)
C_T = 0.002548
CONFIGURATION = CLEAN - ALL DOORS ON
0.99 MAX NAMPP
V_{NE}
SPECIFIC RANGE ~ NAUTICAL AIRMILES PER POUND OF FUEL
0.8
0.6
0.4
0.2
0
CURVE BASED ON SPEC FUEL FLOW FIG 72
MILITARY RATED POWER AVAILABLE
V_{NE}
NORMAL RATED POWER AVAILABLE
RECOMMENDED CRUISE SPEED
CURVE DERIVED FROM FIGS 23 THRU 25
ENGINE OUTPUT SHAFT HORSEPOWER
300
280
260
240
220
200
180
160
140
120
100
80
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

FIGURE 34
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 1595 FT	CG STATION	= 106.9 IN (FWD)
GROSS WEIGHT	= 2632 LB	C_T	= 0.002757
ROTOR SPEED	= 354 RPM	CONFIGURATION	= CLEAN - ALL DOORS ON

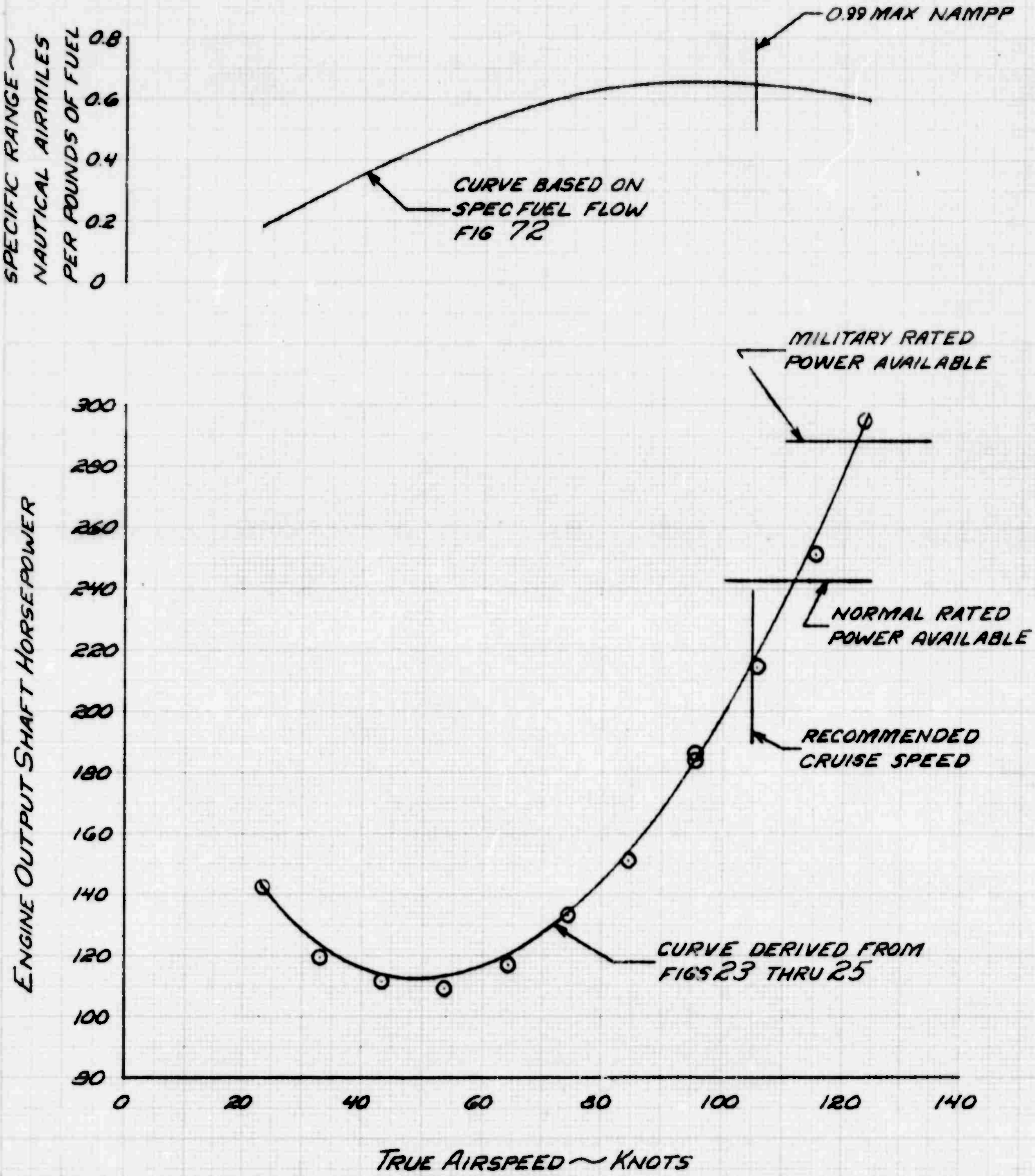


FIGURE 35
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 4175 FT	CG STATION	= 107.2 IN (FWD)
GROSS WEIGHT	= 2752 LB	C_T	= 0.003114
ROTOR SPEED	= 354 RPM	CONFIGURATION	= CLEAN - ALL DOORS ON

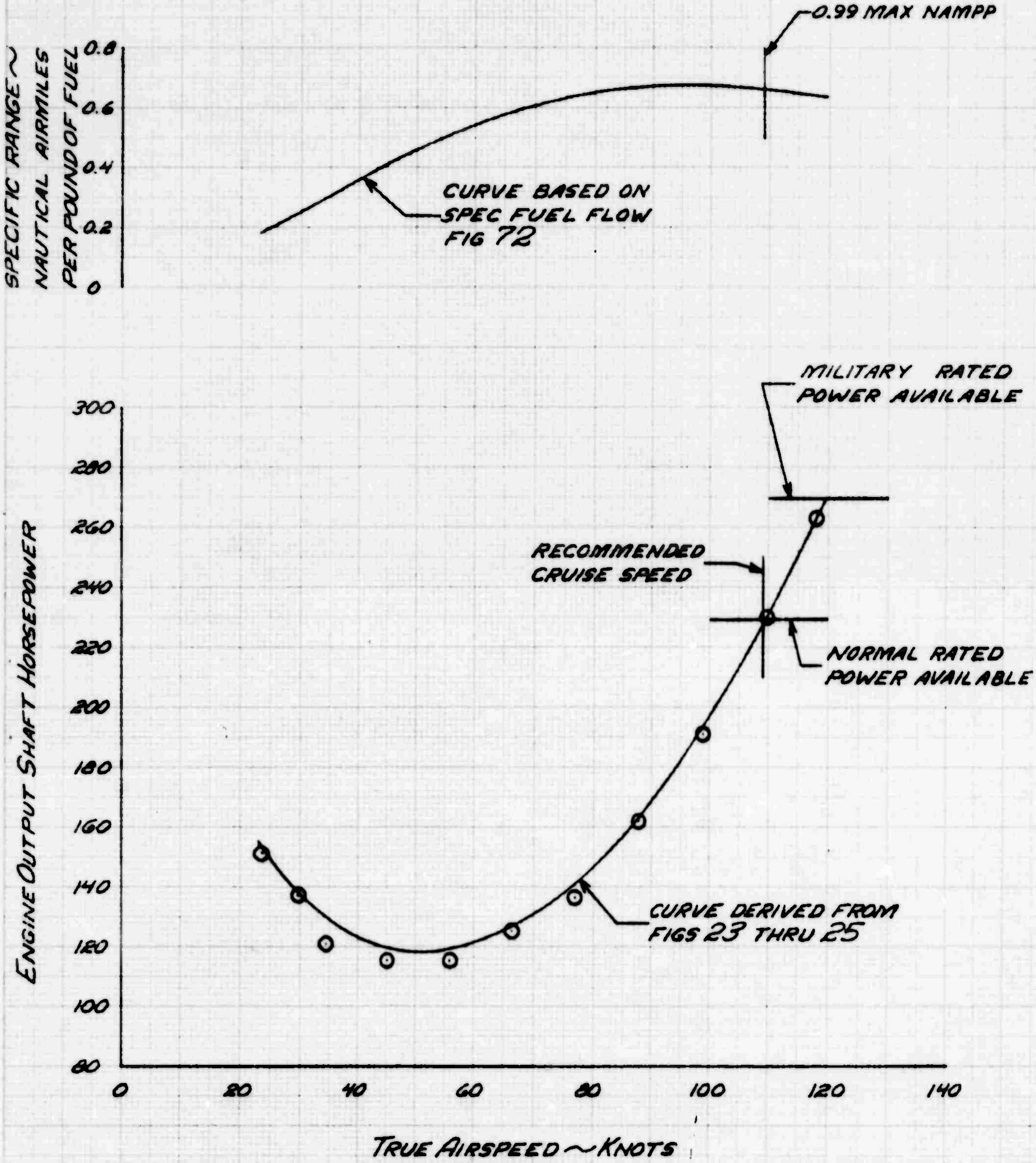
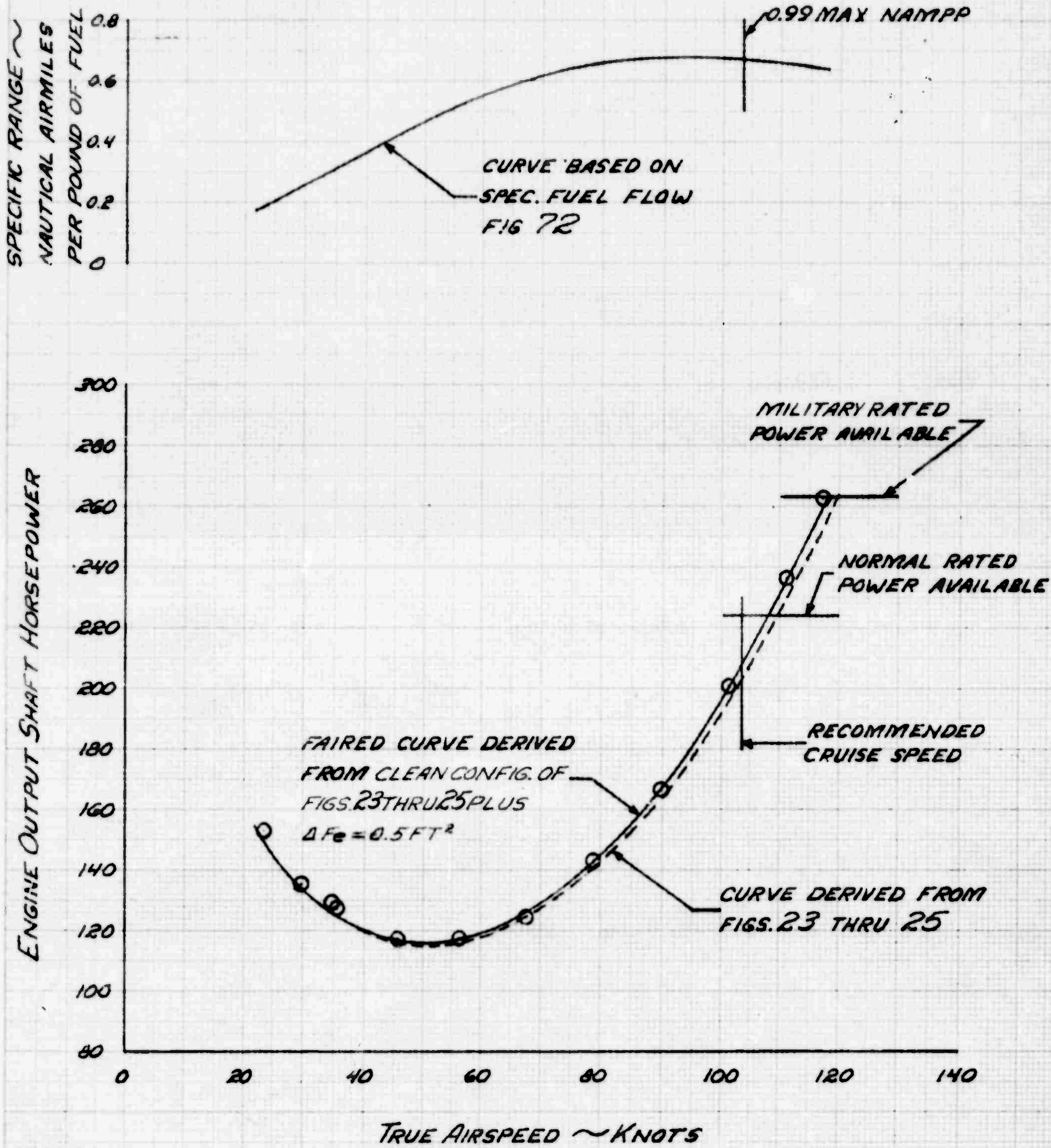


FIGURE 36
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 5090 FT	CG STATION	= 105.9 IN (FWD)
GROSS WEIGHT	= 2678 LB	C_T	= 0.003115
ROTOR SPEED	= 354 RPM	CONFIGURATION	= CLEAN – ALL DOORS ON



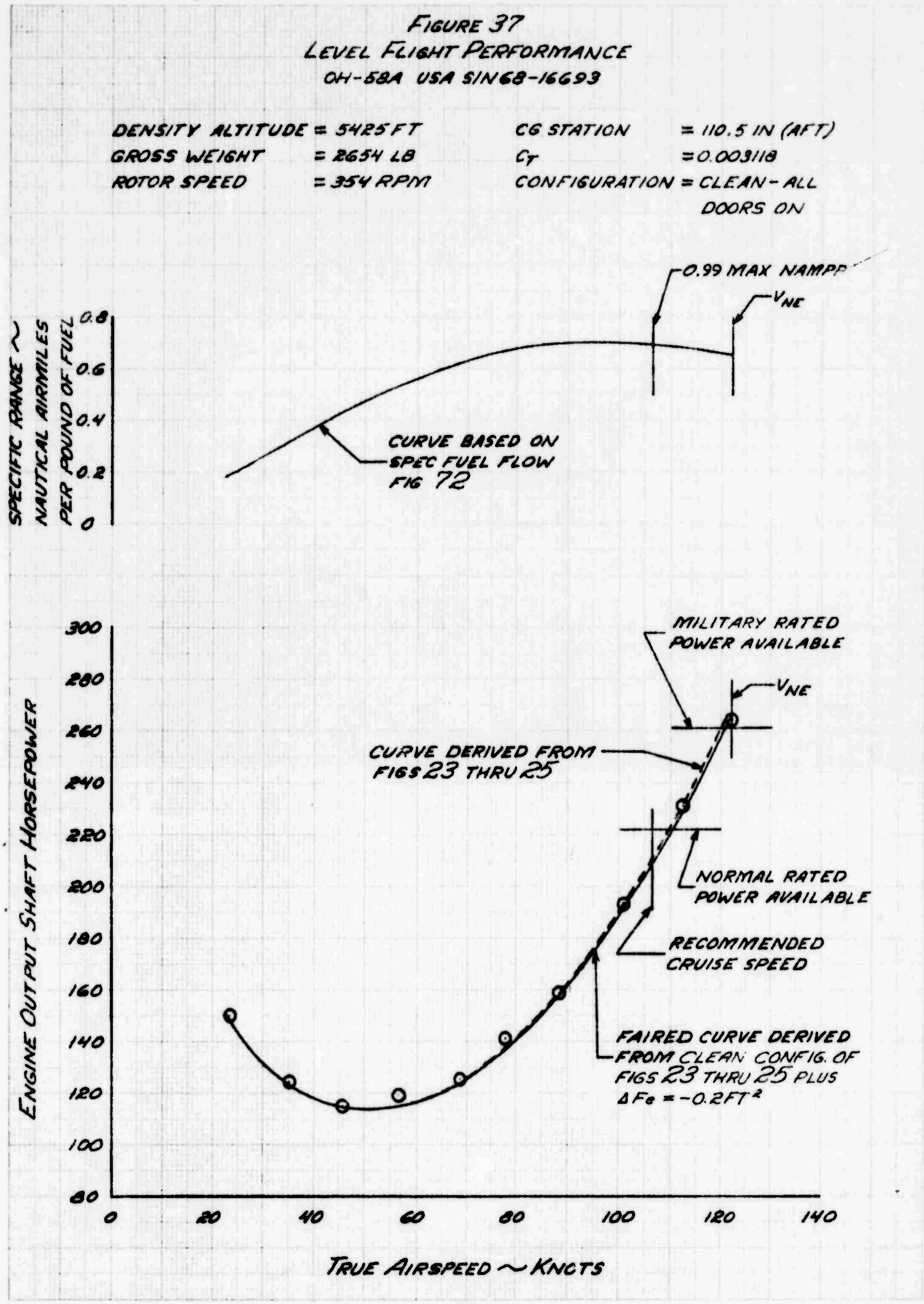


FIGURE 37
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N68-16693
DENSITY ALTITUDE = 5425 FT
GROSS WEIGHT = 2654 LB
ROTOR SPEED = 354 RPM
CG STATION = 110.5 IN (AFT)
C_T = 0.003118
CONFIGURATION = CLEAN - ALL DOORS ON
0.99 MAX NAMPP
V_{NE}
SPECIFIC RANGE ~ NAUTICAL AIRMILES PER POUND OF FUEL
0.8
0.6
0.4
0.2
0
CURVE BASED ON SPEC FUEL FLOW FIG 72
MILITARY RATED POWER AVAILABLE
V_{NE}
CURVE DERIVED FROM FIGS 23 THRU 25
NORMAL RATED POWER AVAILABLE
RECOMMENDED CRUISE SPEED
FAIRED CURVE DERIVED FROM CLEAN CONFIG. OF FIGS 23 THRU 25 PLUS $\Delta F_e = -0.2 FT^2$
ENGINE OUTPUT SHAFT HORSEPOWER
300
280
260
240
220
200
180
160
140
120
100
80
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

FIGURE 38
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

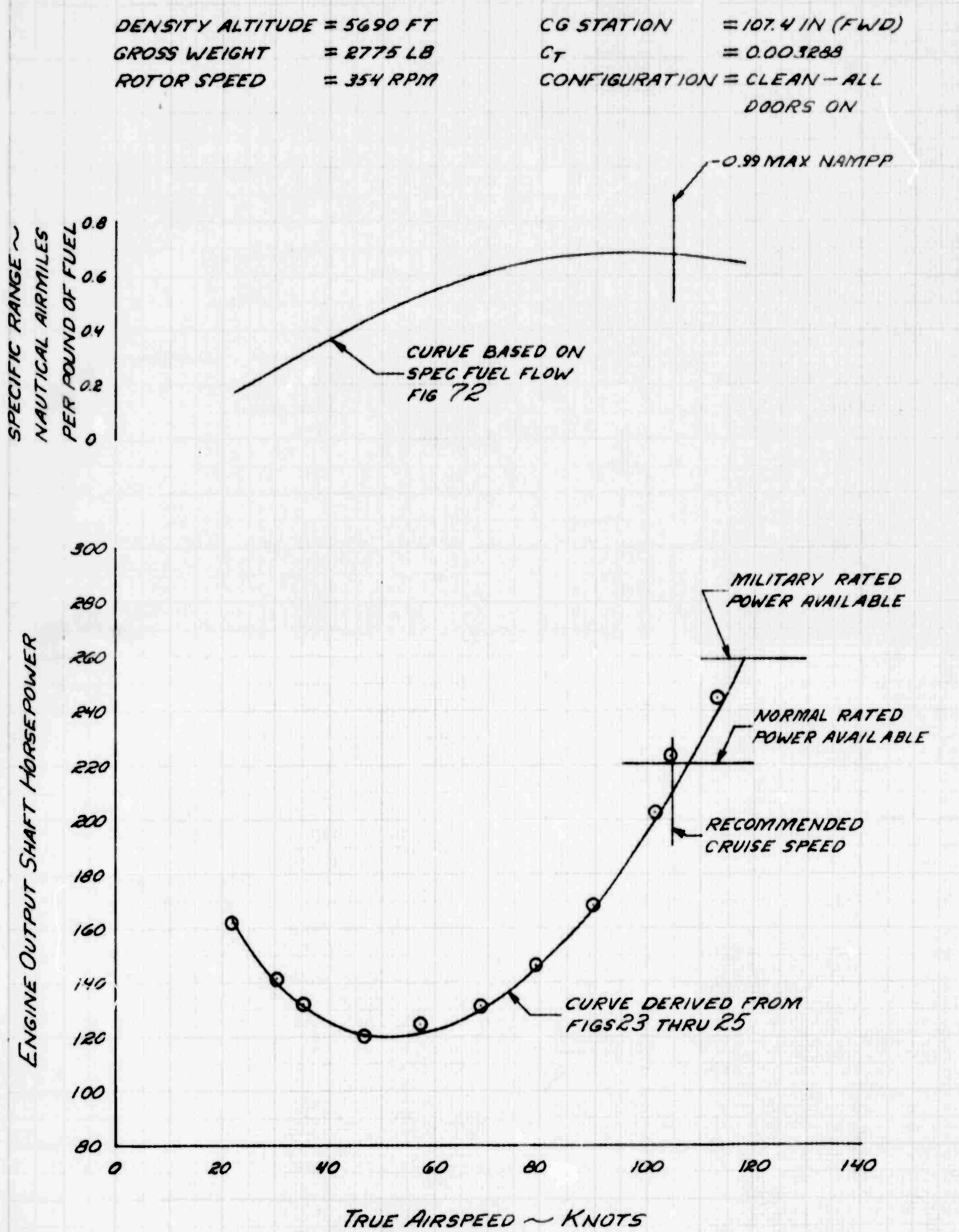


FIGURE 39
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

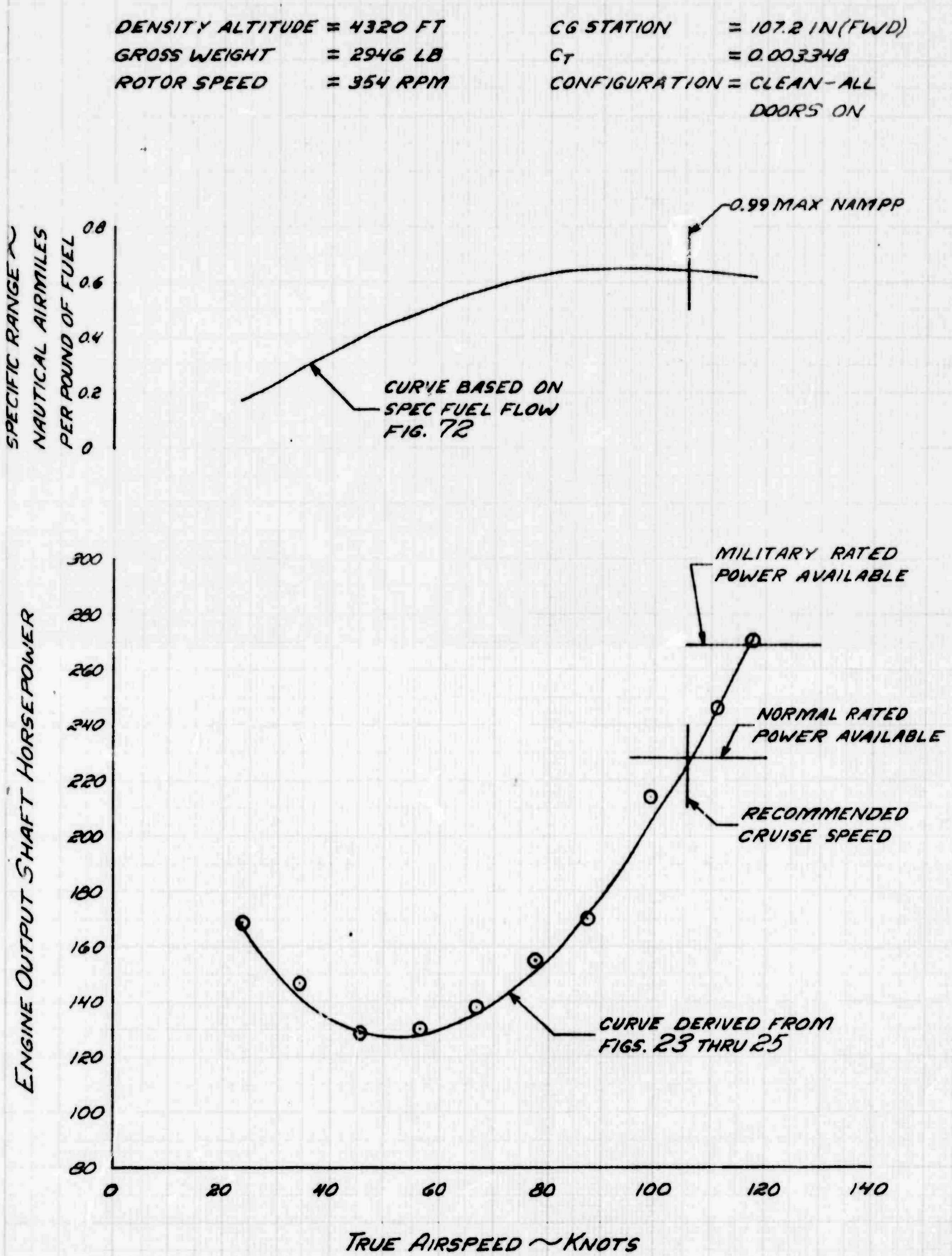


FIGURE 40
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE = 5310 FT
GROSS WEIGHT = 2846 LB
ROTOR SPEED = 347 RPM
CG STATION = 106.7 IN (FWD)
C_T = 0.003469
CONFIGURATION = CLEAN - ALL DOORS ON

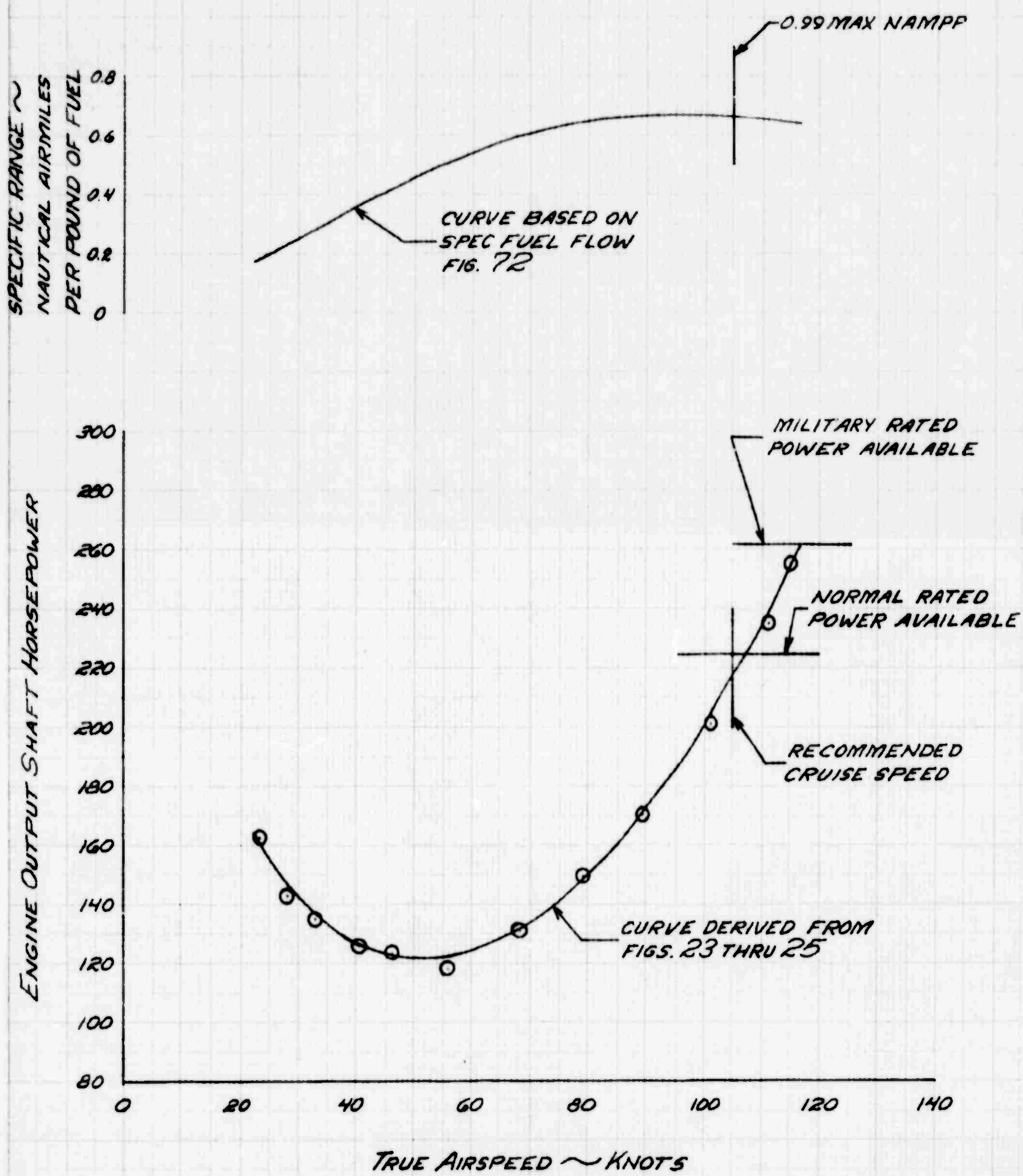


FIGURE 41
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 8955 FT	CG STATION	= 107.4 IN (FWD)
GROSS WEIGHT	= 2781 LB	C_T	= 0.003644
ROTOR SPEED	= 354 RPM	CONFIGURATION	= CLEAN - ALL DOORS ON

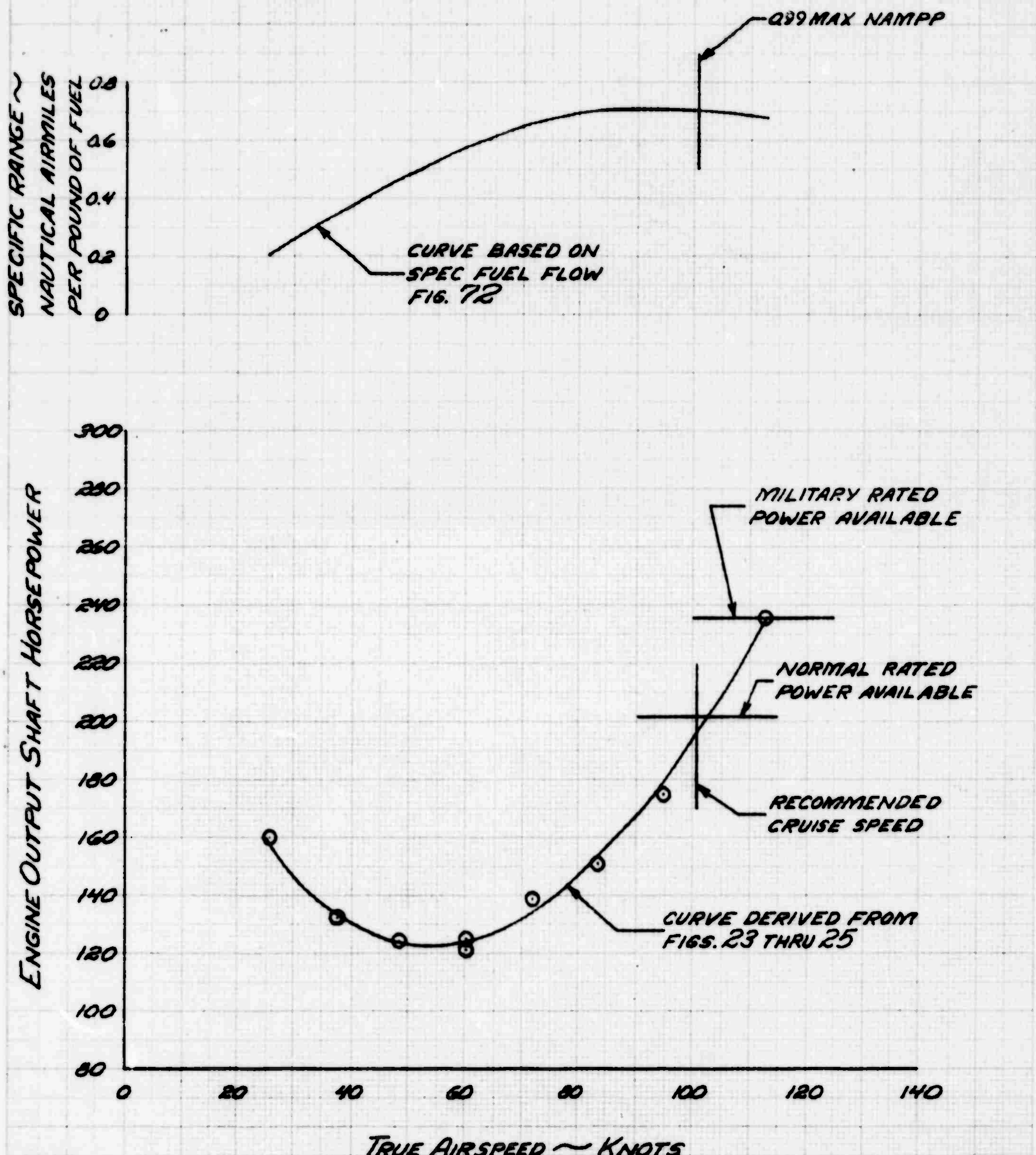


FIGURE 42
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE = 10405 FT
GROSS WEIGHT = 2950 LB
ROTOR SPEED = 354 RPM

CG STATION = 106.8 IN (FWD)
C_T = 0.004045
CONFIGURATION = CLEAN - ALL DOORS ON

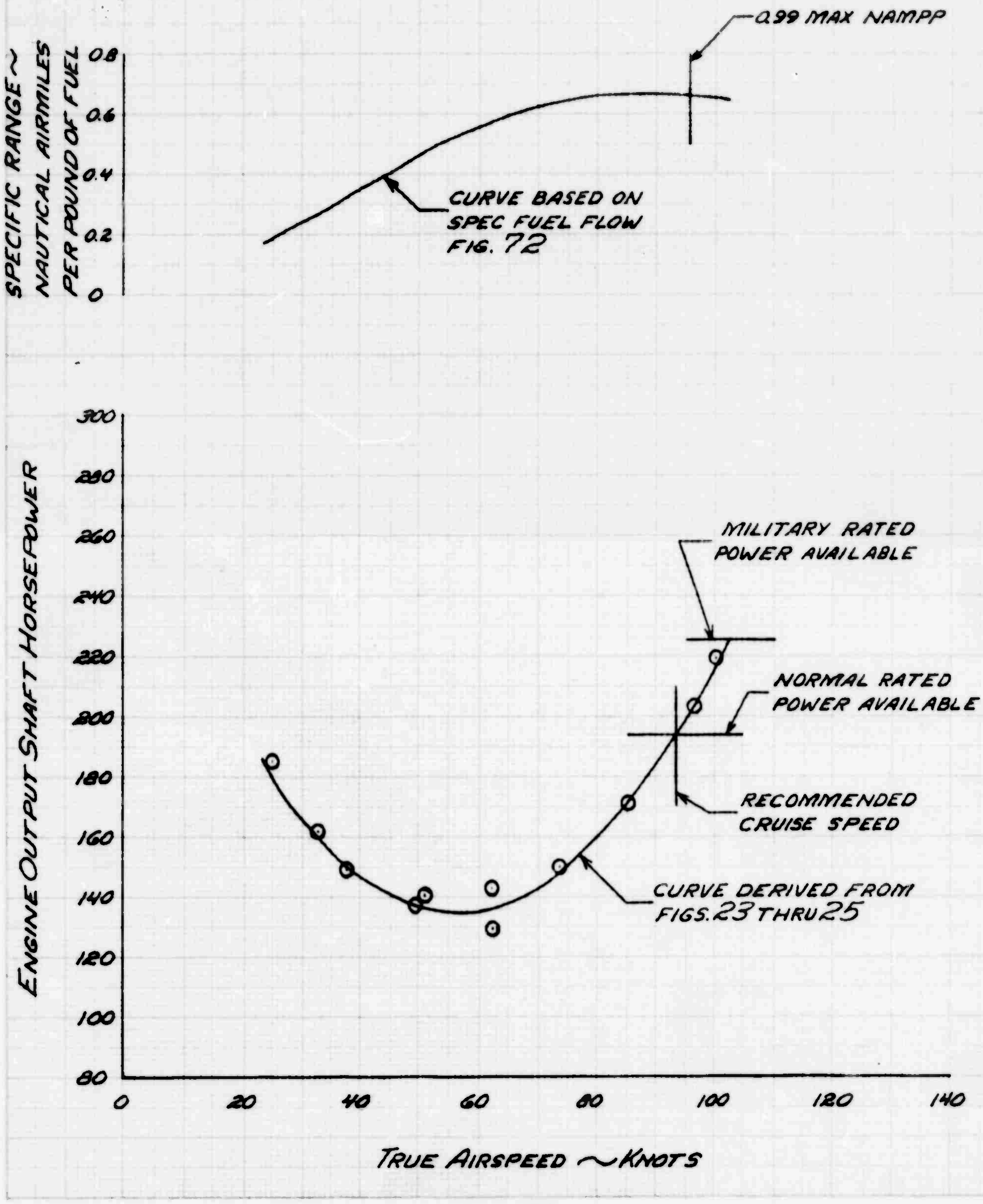


FIGURE 43
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

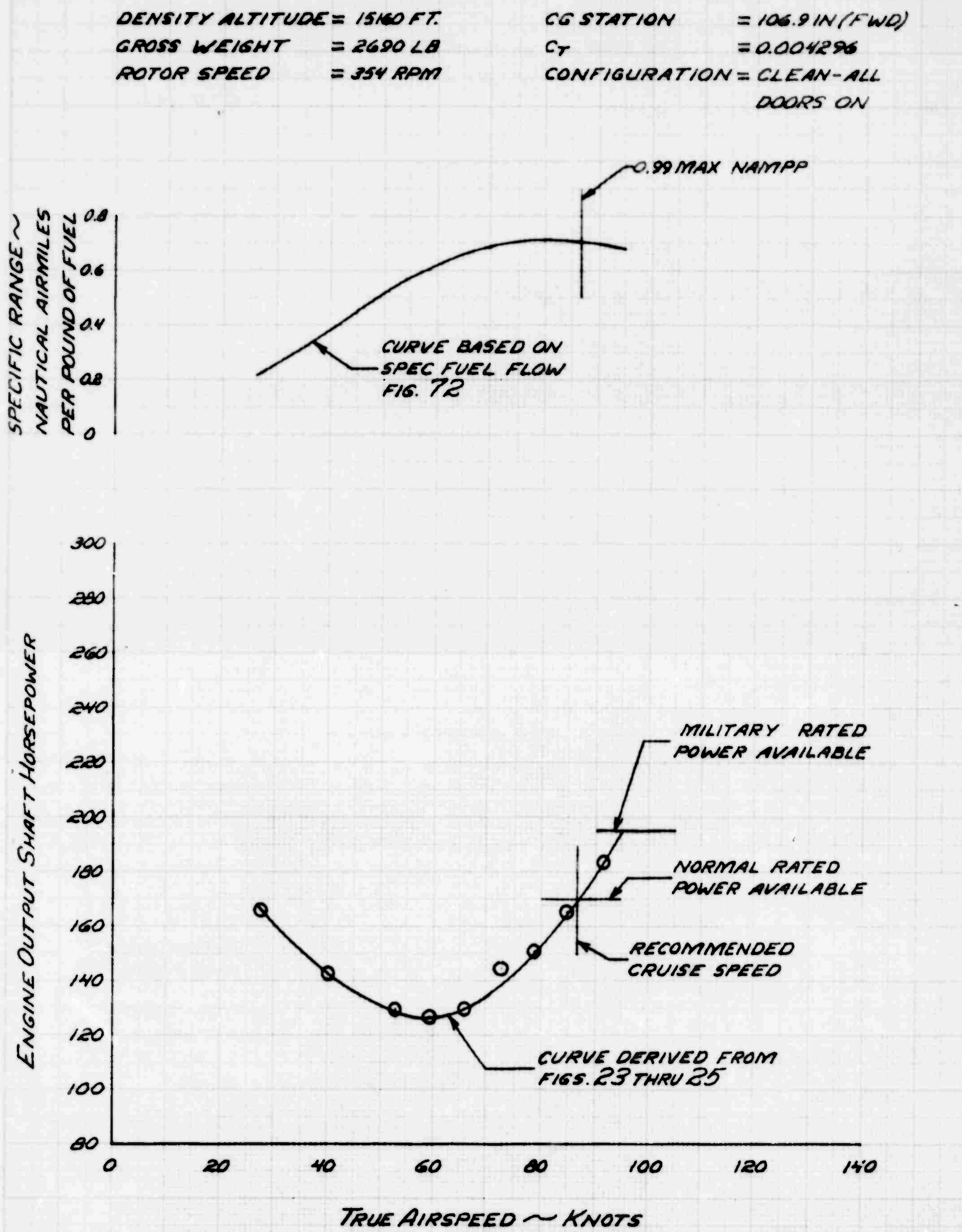


FIGURE 44
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE = 14530 FT.
GROSS WEIGHT = 2224 LB.
ROTOR SPEED = 354 RPM
CG STATION = 106.9 IN (FWD)
C_T = 0.003480
CONFIGURATION = CLEAN - ALL DOORS ON

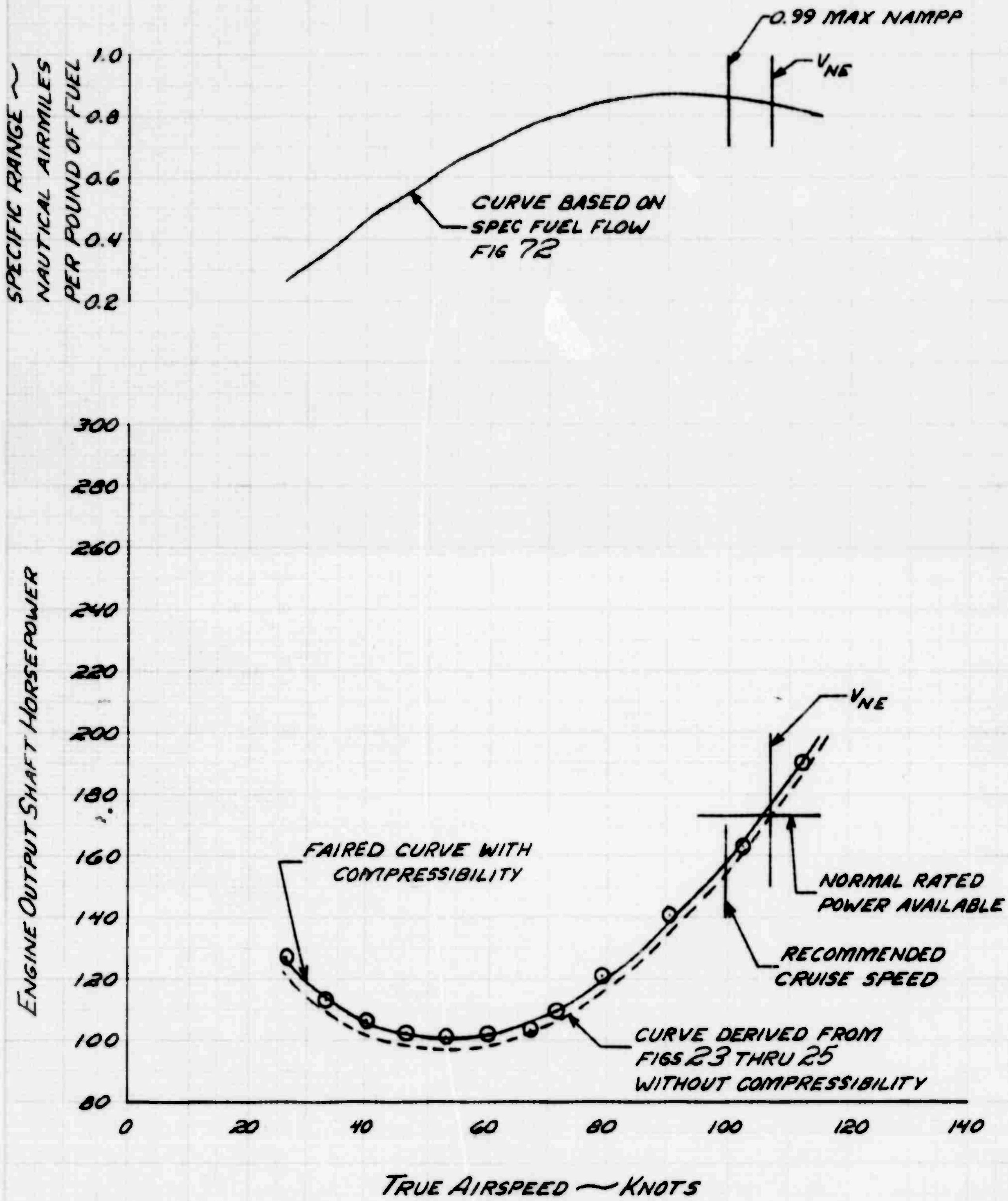


FIGURE 45
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 1490 FT.	CG STATION	= 106.9 IN (FWD)
GROSS WEIGHT	= 2641 LB.	C_T	= 0.002758
ROTOR SPEED	= 354 RPM	CONFIGURATION	= ARMED-ALL DOORS ON

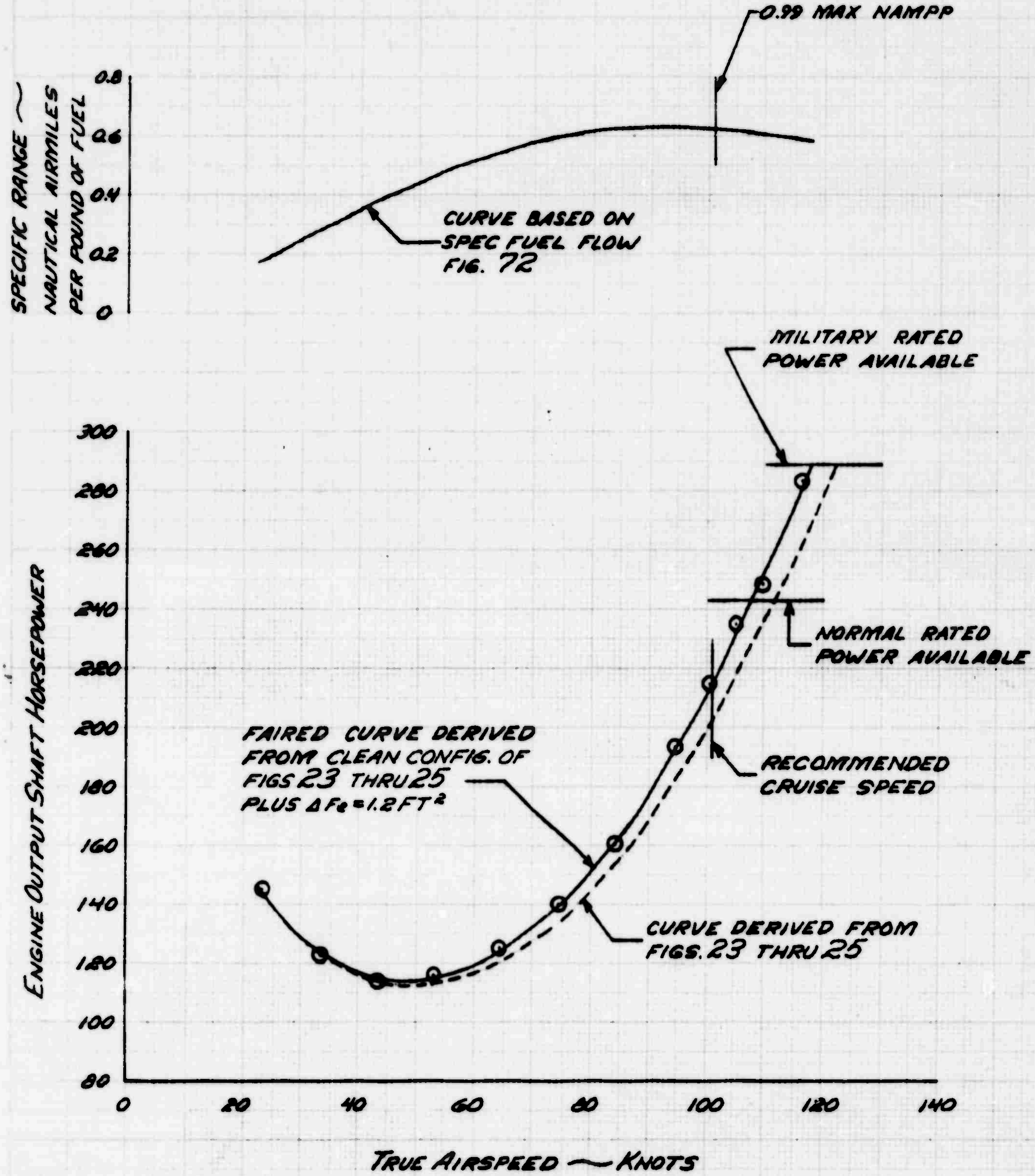


FIGURE 46
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 5055 FT	CG STATION	= 107.0 IN (FWD)
GROSS WEIGHT	= 2694 LB	C_T	= 0.003131
ROTOR SPEED	= 354 RPM	CONFIGURATION	= ARMED-ALL DOORS ON

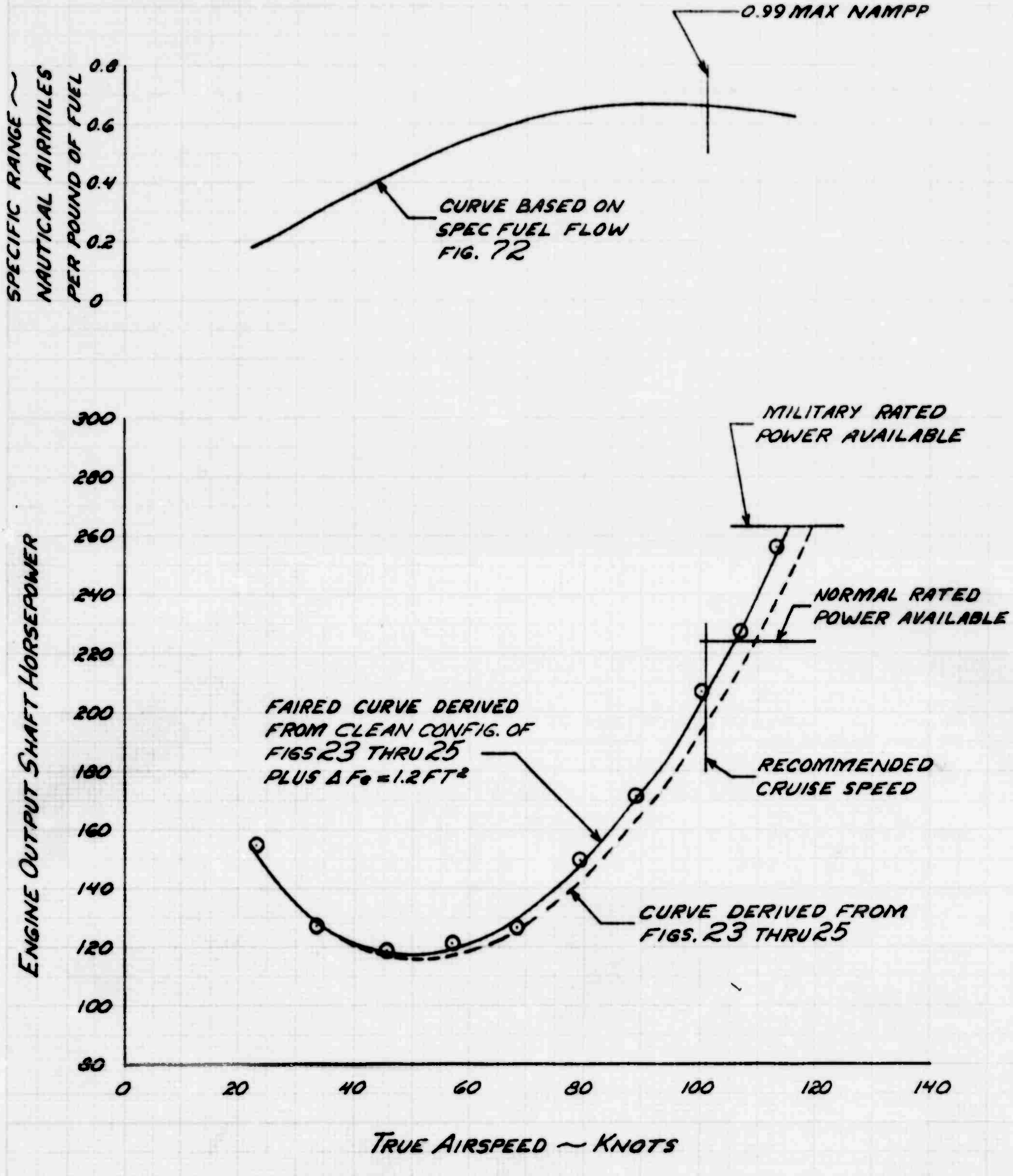


FIGURE 47
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 4815 FT	CG STATION	= 107.1 IN (FWD)
GROSS WEIGHT	= 3007 LB	C_T	= 0.003469
ROTOR SPEED	= 354 RPM	CONFIGURATION	= ARMED - ALL DOORS ON

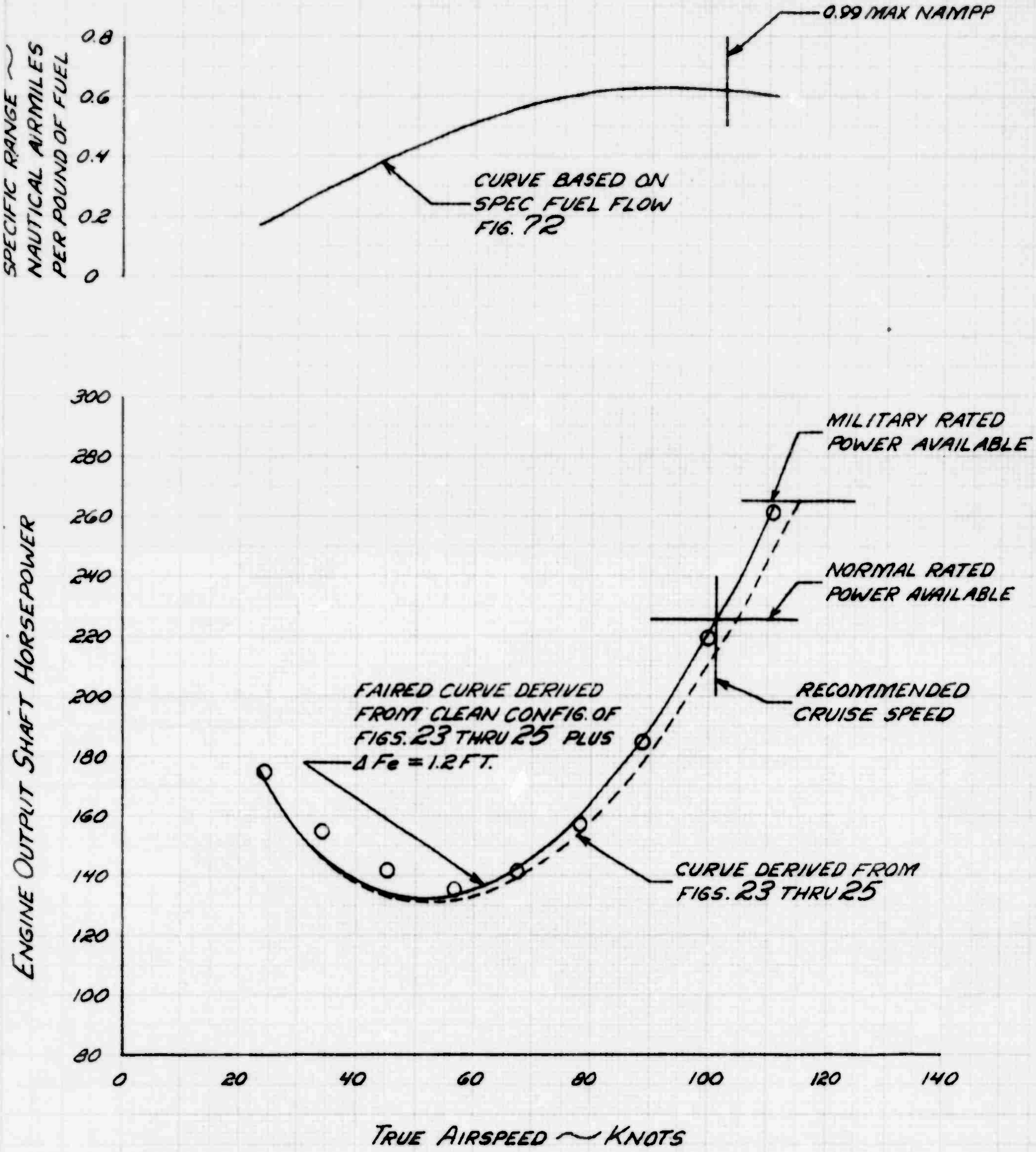


FIGURE 48
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE = 1495 FT
GROSS WEIGHT = 2616 LB
ROTOR RPM = 354 RPM
CG STATION = 106.8 IN. (FWD)
C_T = 0.002733
CONFIGURATION = ARMED-CARGO DOORS OFF

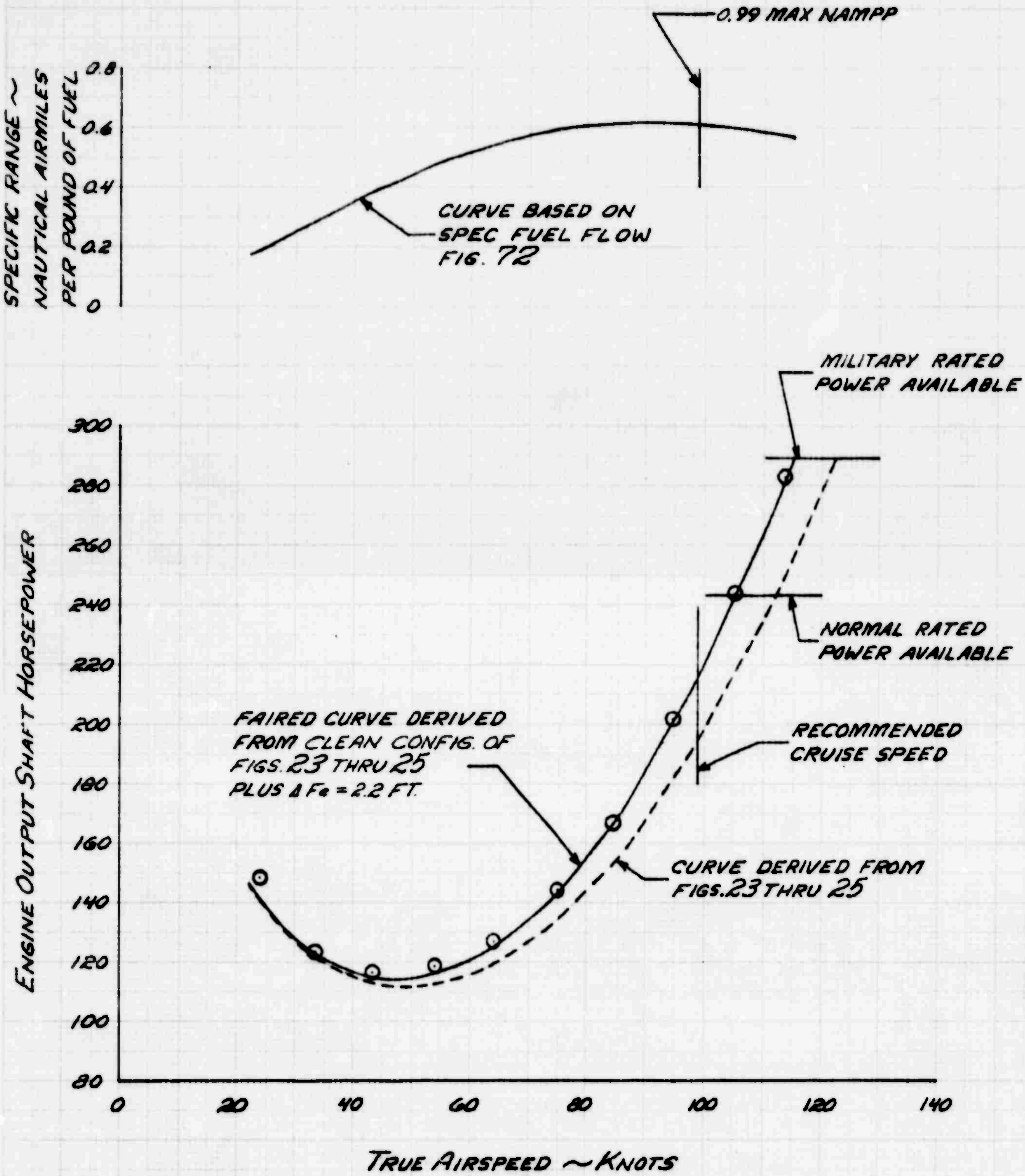


FIGURE 49
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 4580 FT	CG STATION	= 107.1 IN (FWD)
GROSS WEIGHT	= 2720 LB	C_T	= 0.003115
ROTOR SPEED	= 354 RPM	CONFIGURATION	= ARMED-CARGO DOORS OFF

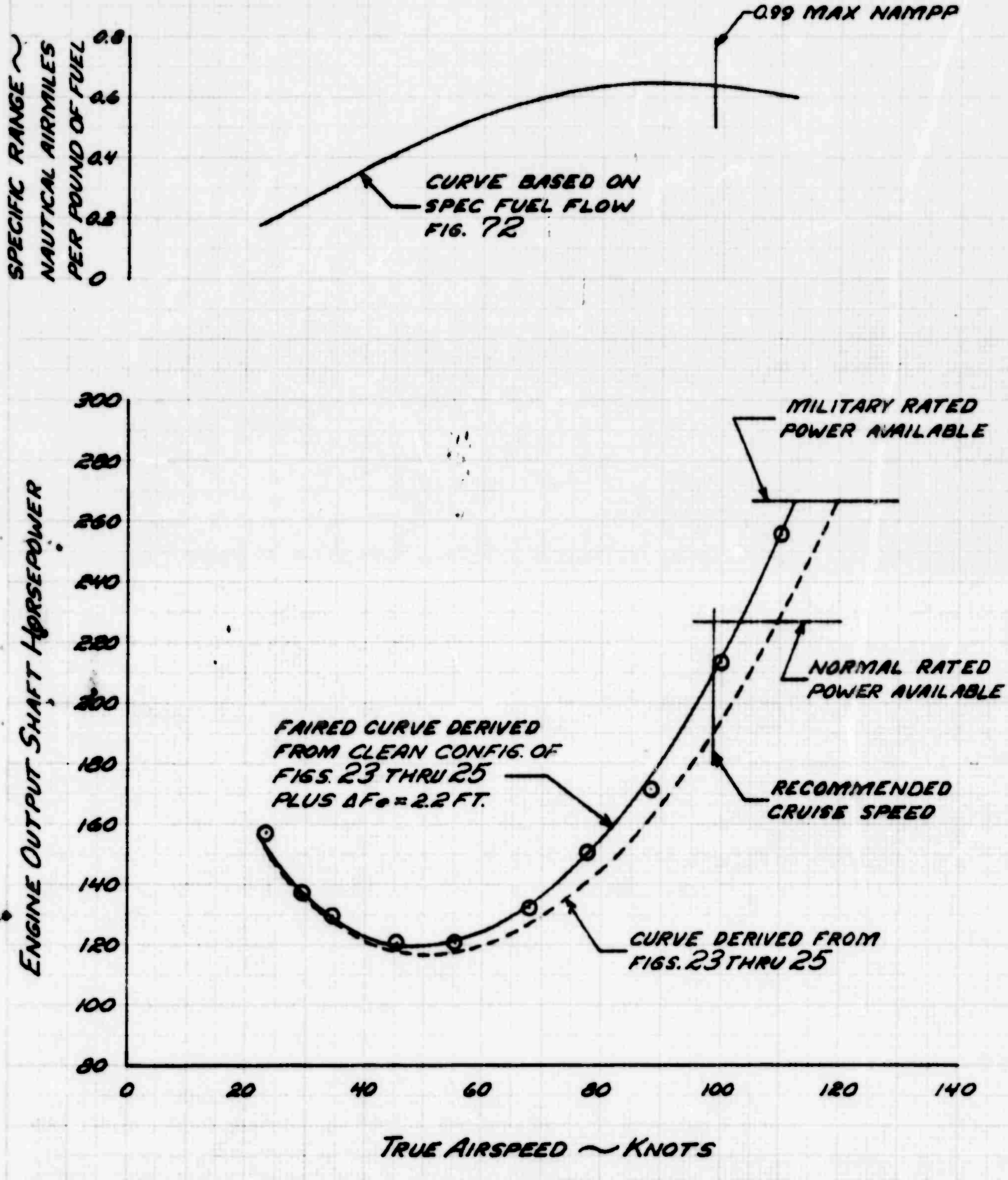


FIGURE 50
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 4795 FT.	CG STATION	= 107.0 IN (FWD)
GROSS WEIGHT	= 2992 LB.	C_T	= 0.003450
ROTOR SPEED	= 354 RPM	CONFIGURATION	= ARMED-CARGO DOORS OFF

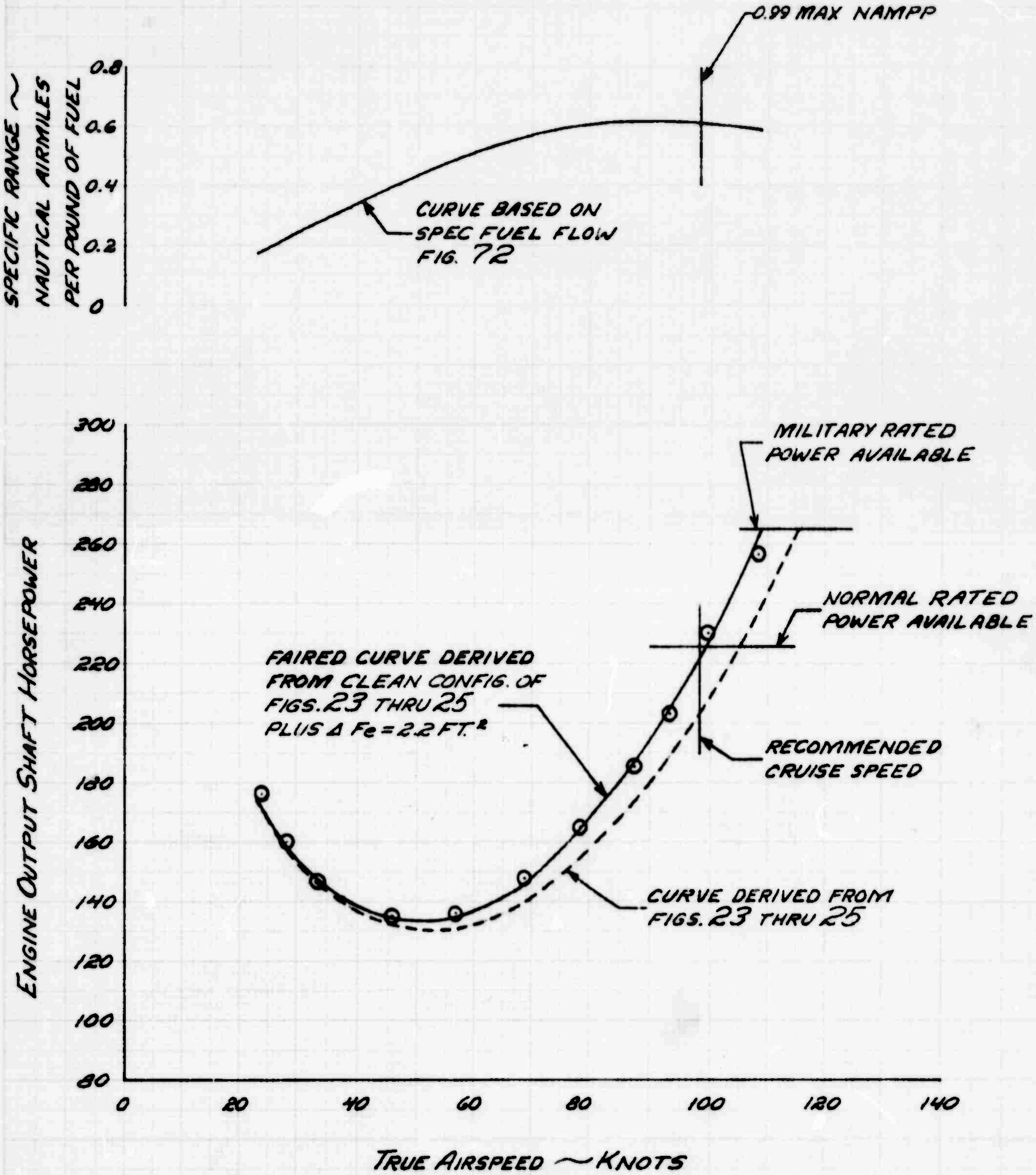


FIGURE 51
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

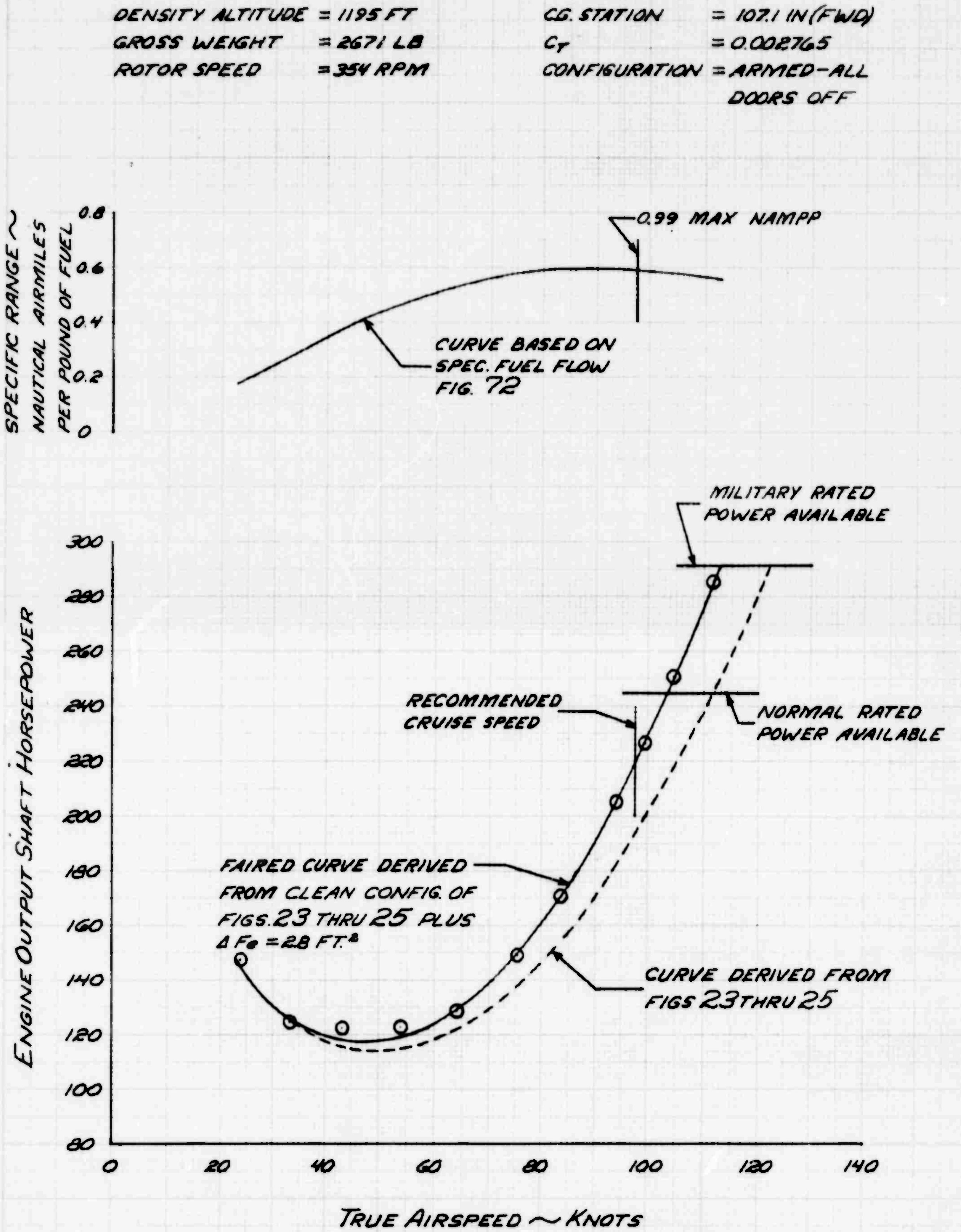


FIGURE 52
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

DENSITY ALTITUDE	= 4740 FT	CG STATION	= 107.1 IN (FWD)
GROSS WEIGHT	= 2713 LB	C_T	= 0.003123
ROTOR SPEED	= 354 RPM	CONFIGURATION	= ARMED - ALL DOORS OFF

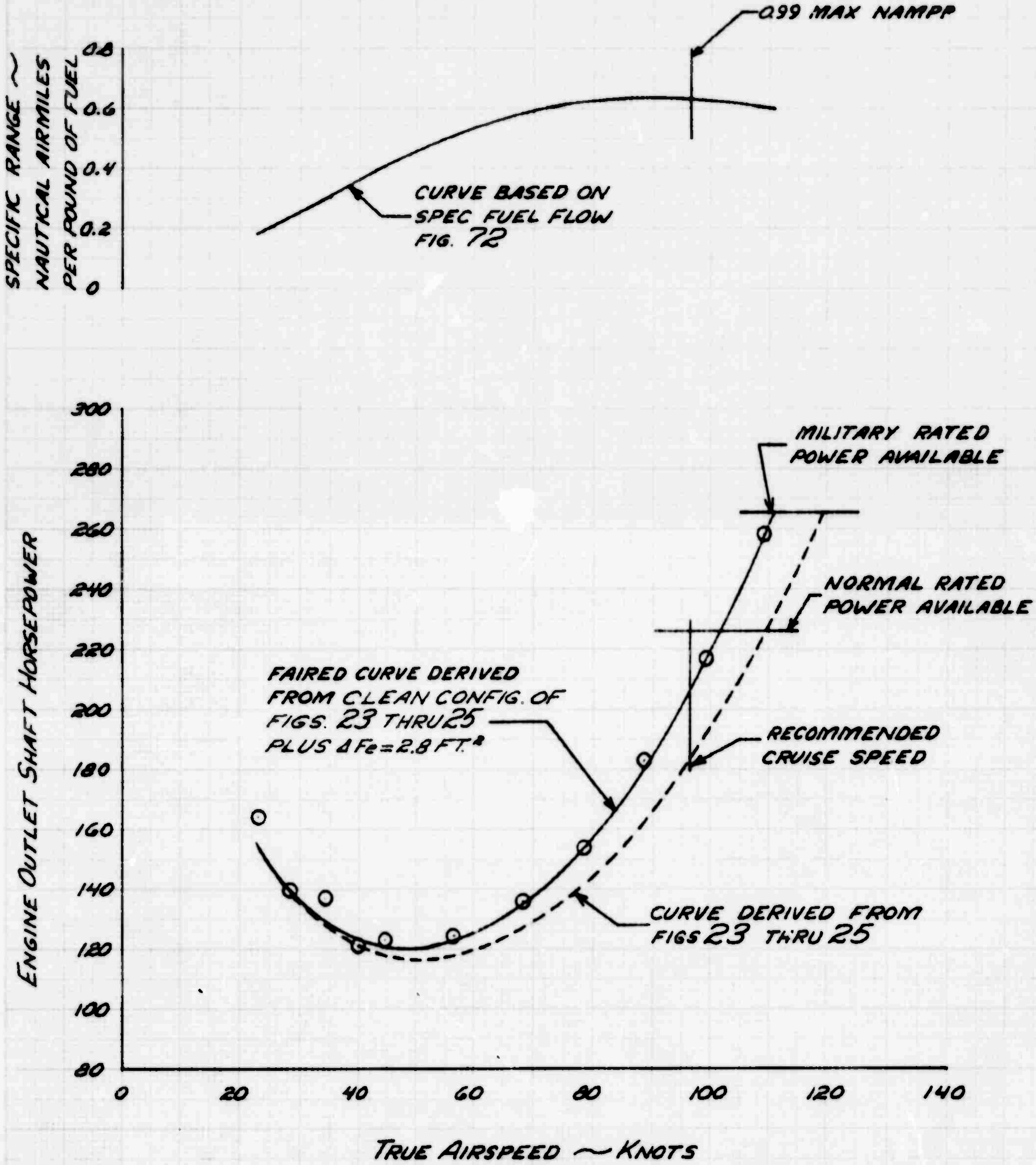


FIGURE 53
LEVEL FLIGHT PERFORMANCE
OH-58A USA S/N 68-16693

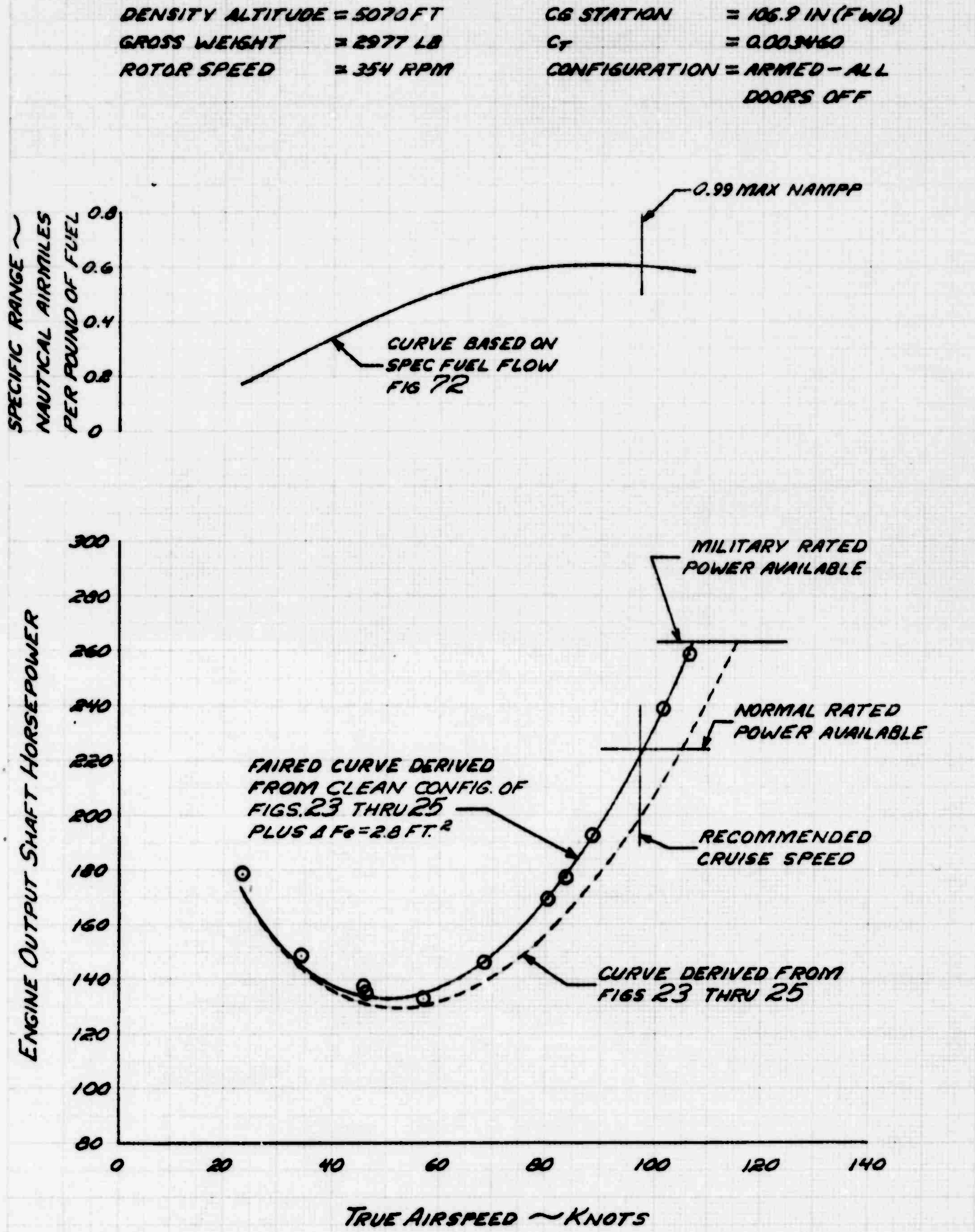


FIGURE 54
AUTOROTATIONAL DESCENTS
OH-58A USA S/N 68-16693
CLEAN CONFIGURATION

NOTE: 1 CURVES OBTAINED FROM FIGS 55 THRU 57

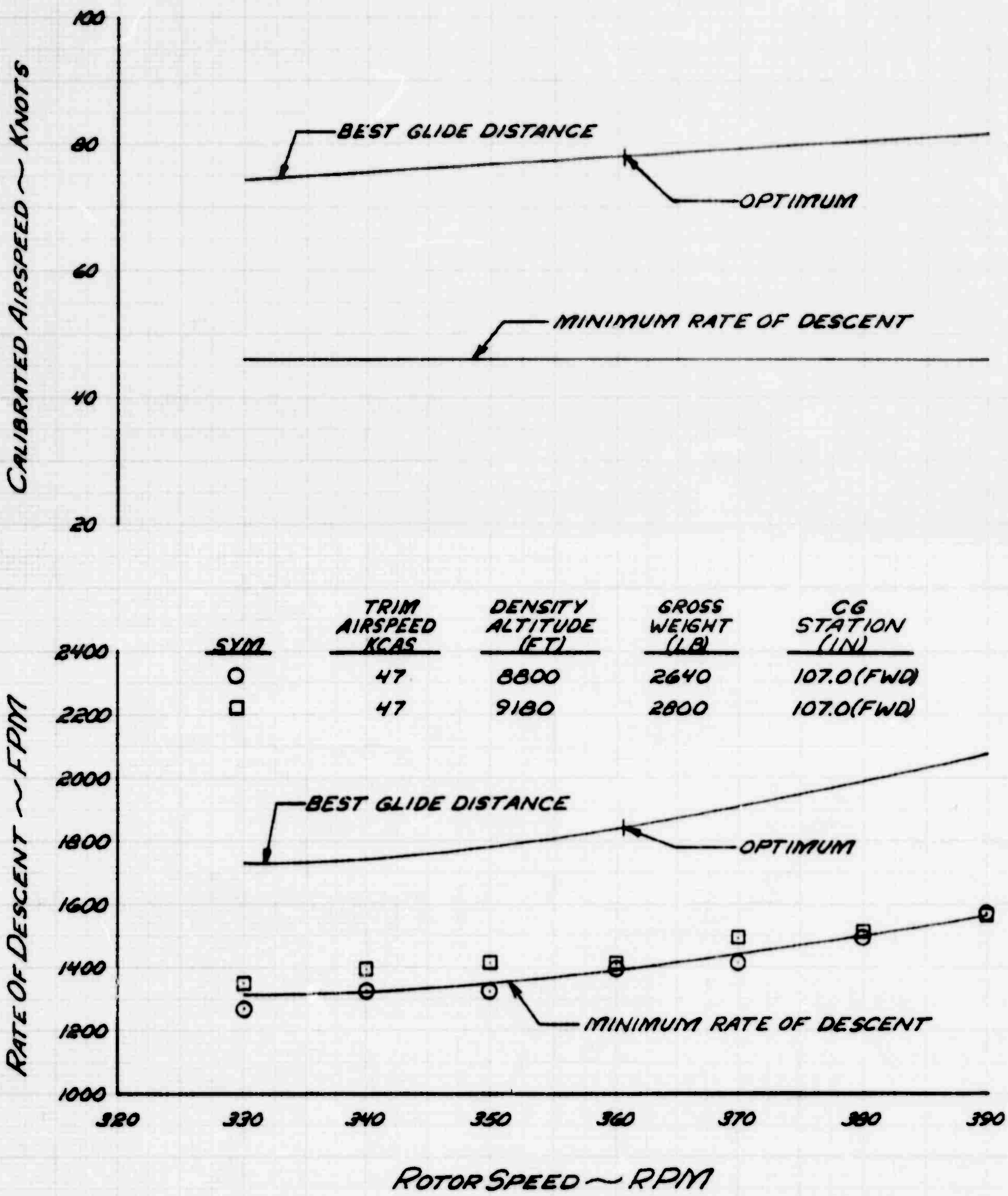


FIGURE 55
AUTOROTATIONAL DESCENTS
OH-58A USA S/N 68-16693
CLEAN CONFIGURATION
330 RPM

SYM	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	CG STATION (IN)
○	9590	2640	107.0(FWD)
□	8930	2800	107.0(FWD)
△	12150	2160	107.0(FWD)

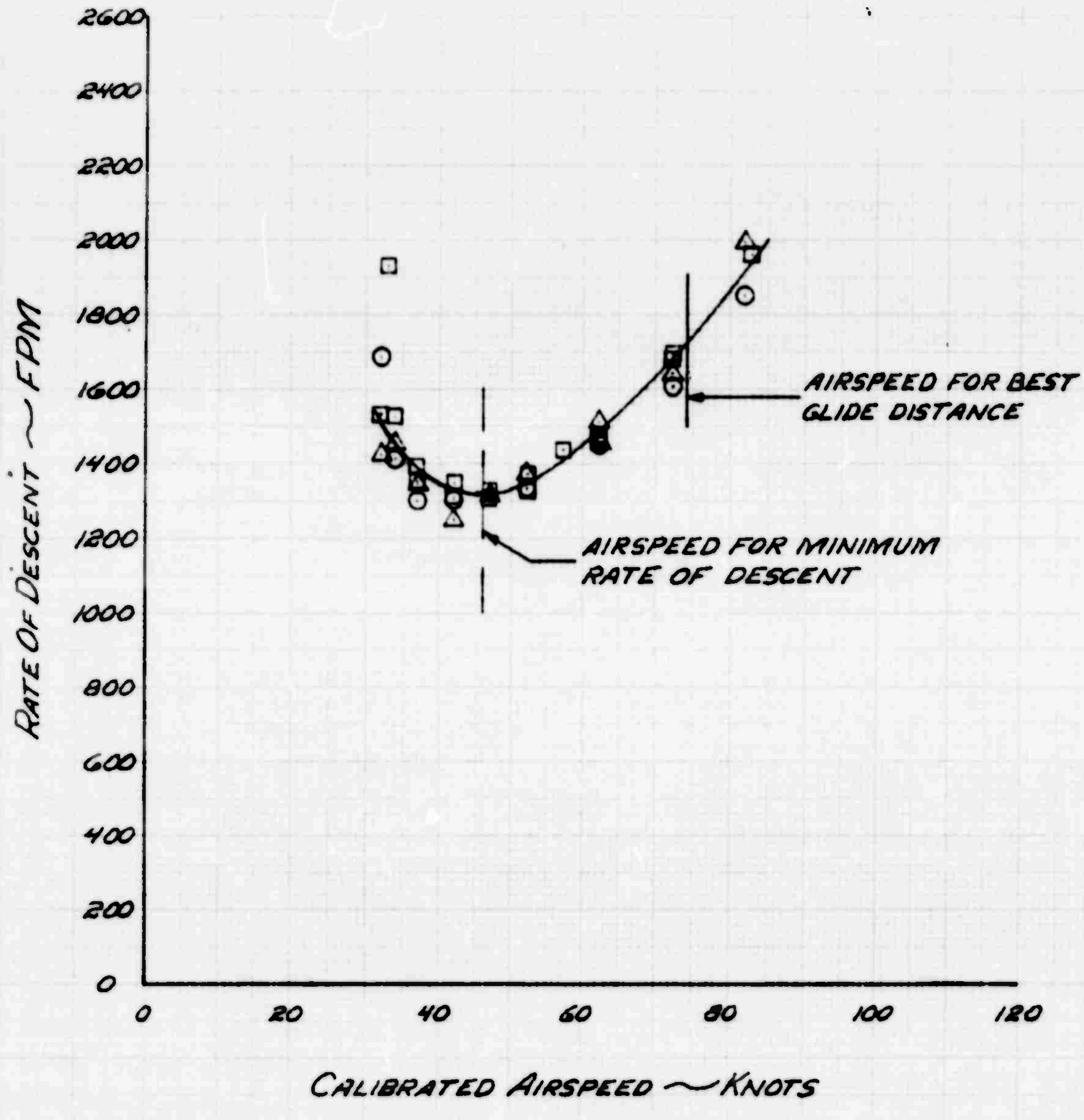


FIGURE 56

AUTOROTATIONAL DESCENTS

OH-58A USA S/N 68-16693

CLEAN CONFIGURATION

354 RPM

SYM	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	CG STATION (IN)
○	9195	2640	107.0 (FWD)
□	10590	2800	107.0 (FWD)
△	12360	2160	107.0 (FWD)

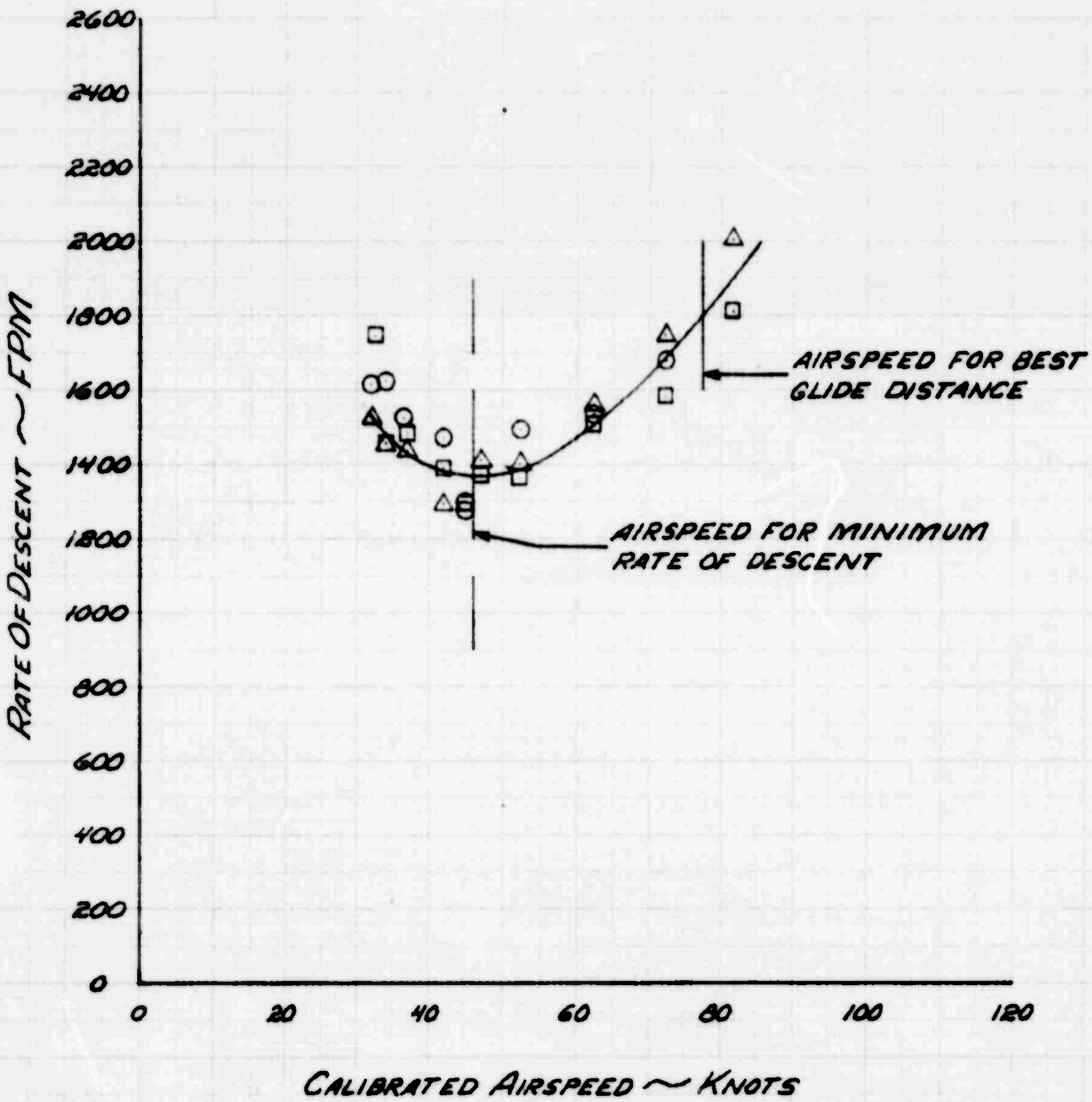


FIGURE 57
AUTOROTATIONAL DESCENTS
OH-58A USA S/N 68-16693
CLEAN CONFIGURATION
390 RPM

SYM	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	CG STATION (IN)
○	13265	2640	107.0 (FWD)
□	10385	2800	107.0 (FWD)
△	12690	2160	107.0 (FWD)

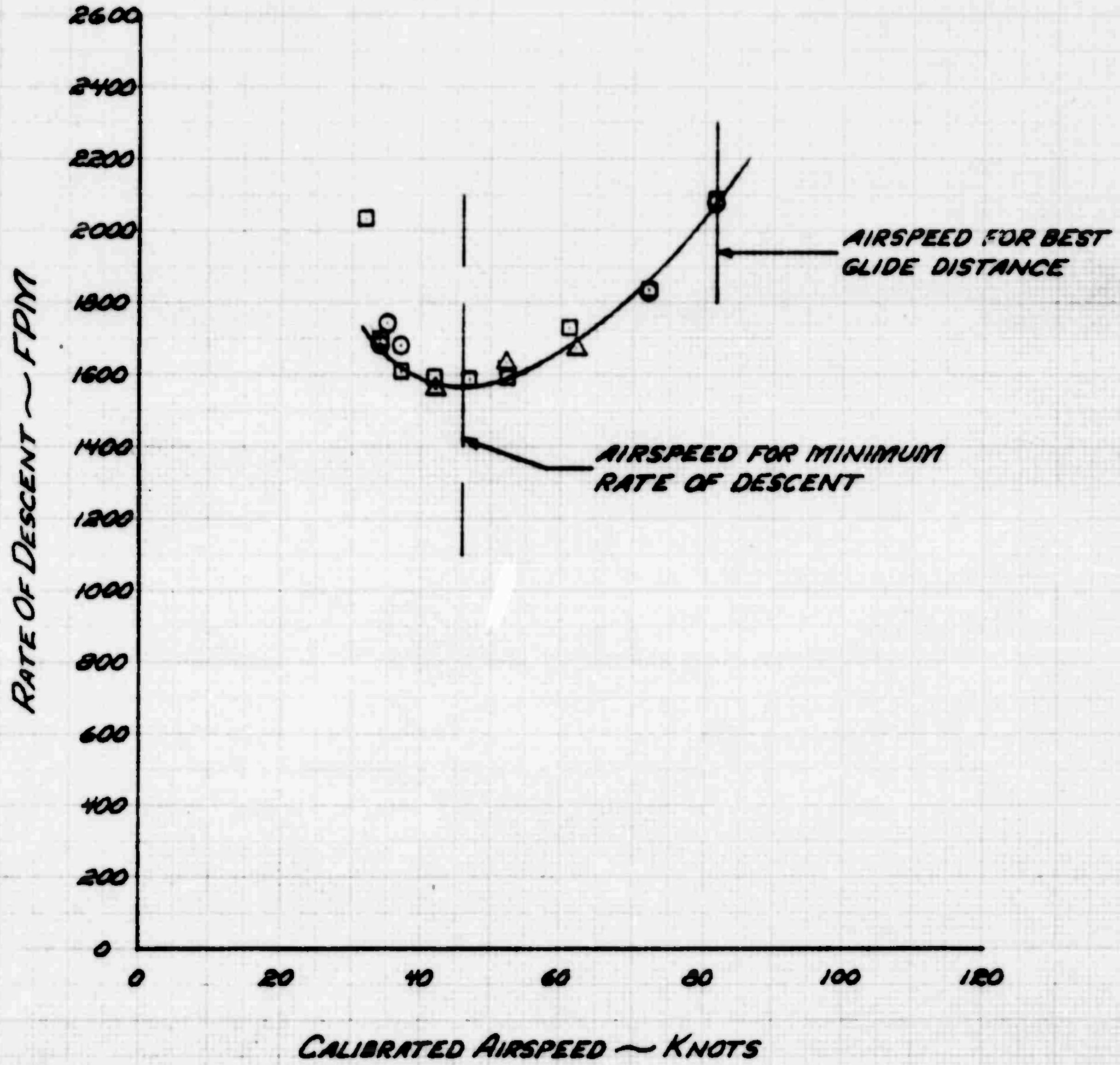


FIGURE 58
COMPRESSOR INLET CHARACTERISTIC COMPARISON
OH-58A USA S/N 68-16693
T63-A-700 ENGINE

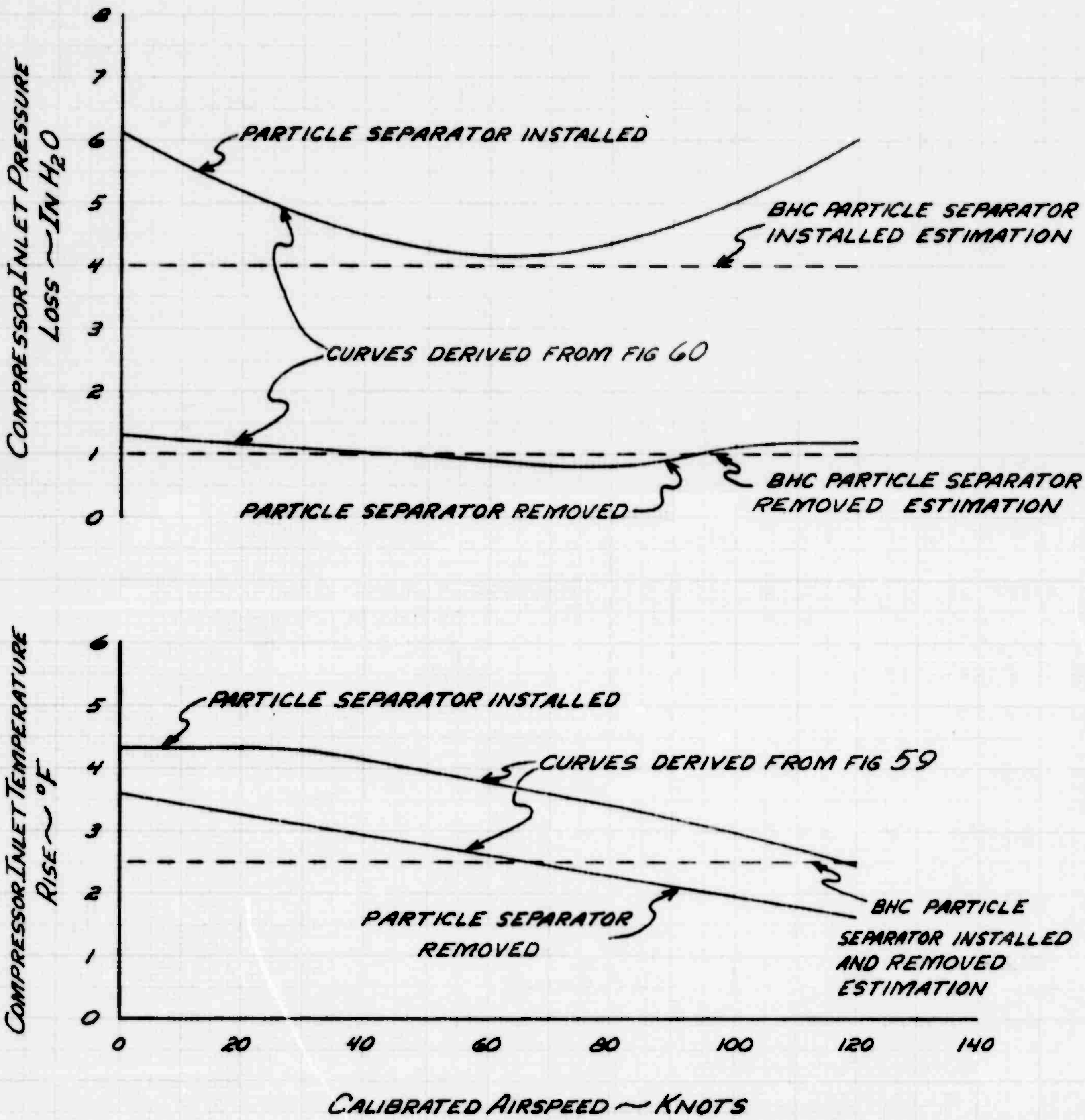


Figure 59
Engine Inlet Characteristics
OH-58A USA S/N 68-16693
T63-A-700 Engine

SYM	PRESS ALT (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)	CG STATION
○	2995	2400	354	107.0 (FWD)
□	5960	2400	354	107.0(FWD)
△	510	2400	354	107.0(FWD)
◇	9930	2400	354	107.0(FWD)
⬡	1015	2400	354	107.0 (FWD)
◸	4970	2400	354	107.0(FWD)
△	9955	2400	354	107.0(FWD)

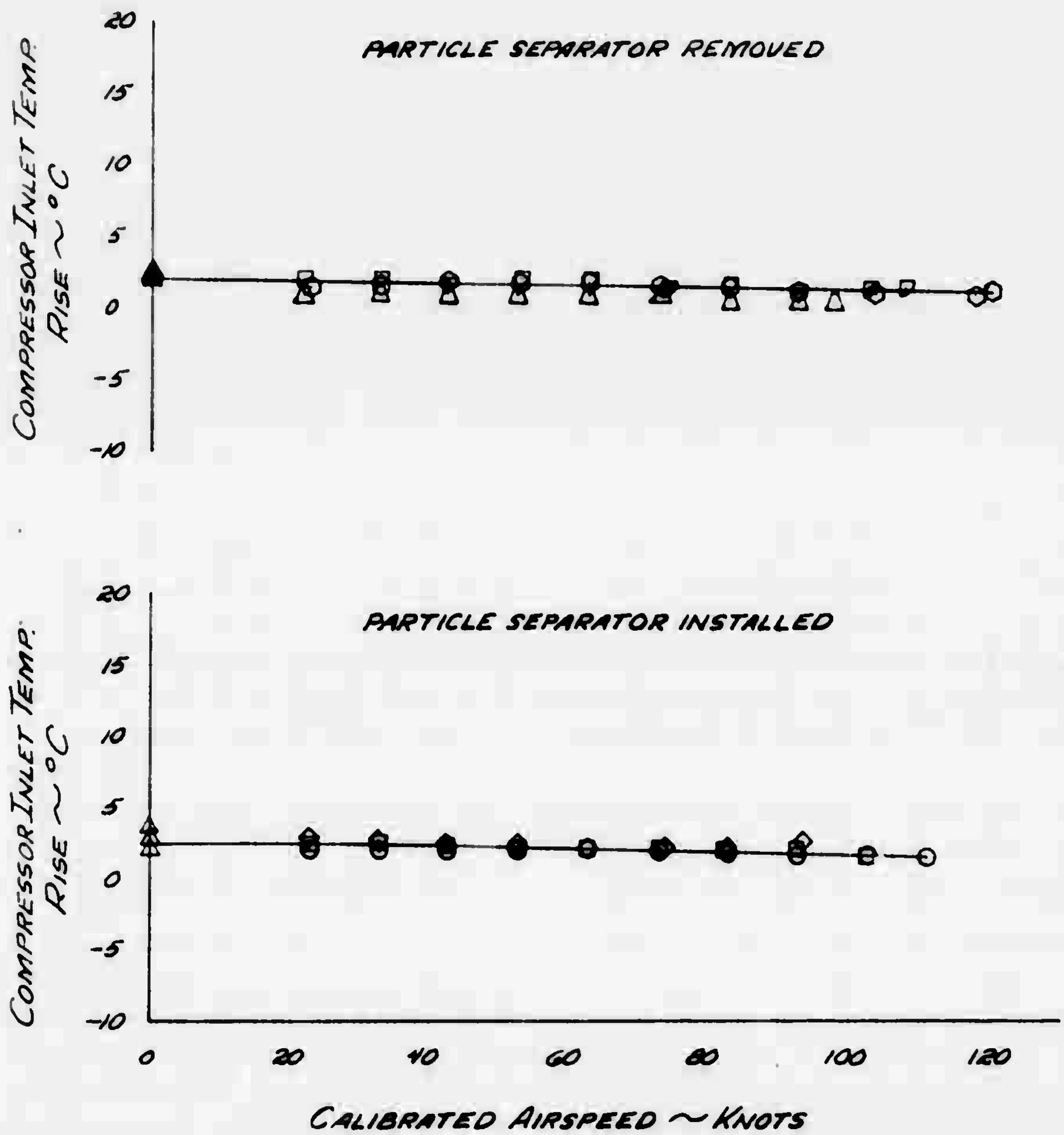


FIGURE 60

ENGINE INLET CHARACTERISTICS

OH-58A USA S/N 68-16693

T63-A-700 ENGINE

SYM	PRESS ALT (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)	CG STATION
○	2995	2400	354	107.0 (FWD)
□	5960	2400	354	107.0 (FWD)
△	9930	2400	354	107.0 (FWD)
◇	510	2400	354	107.0 (FWD)
⬭	840	2630	354	106.9 (FWD)
⬯	1140	2180	354	107.0 (FWD)
○	1015	2400	354	107.0 (FWD)
□	4970	2400	354	107.0 (FWD)
△	9995	2400	354	107.0 (FWD)

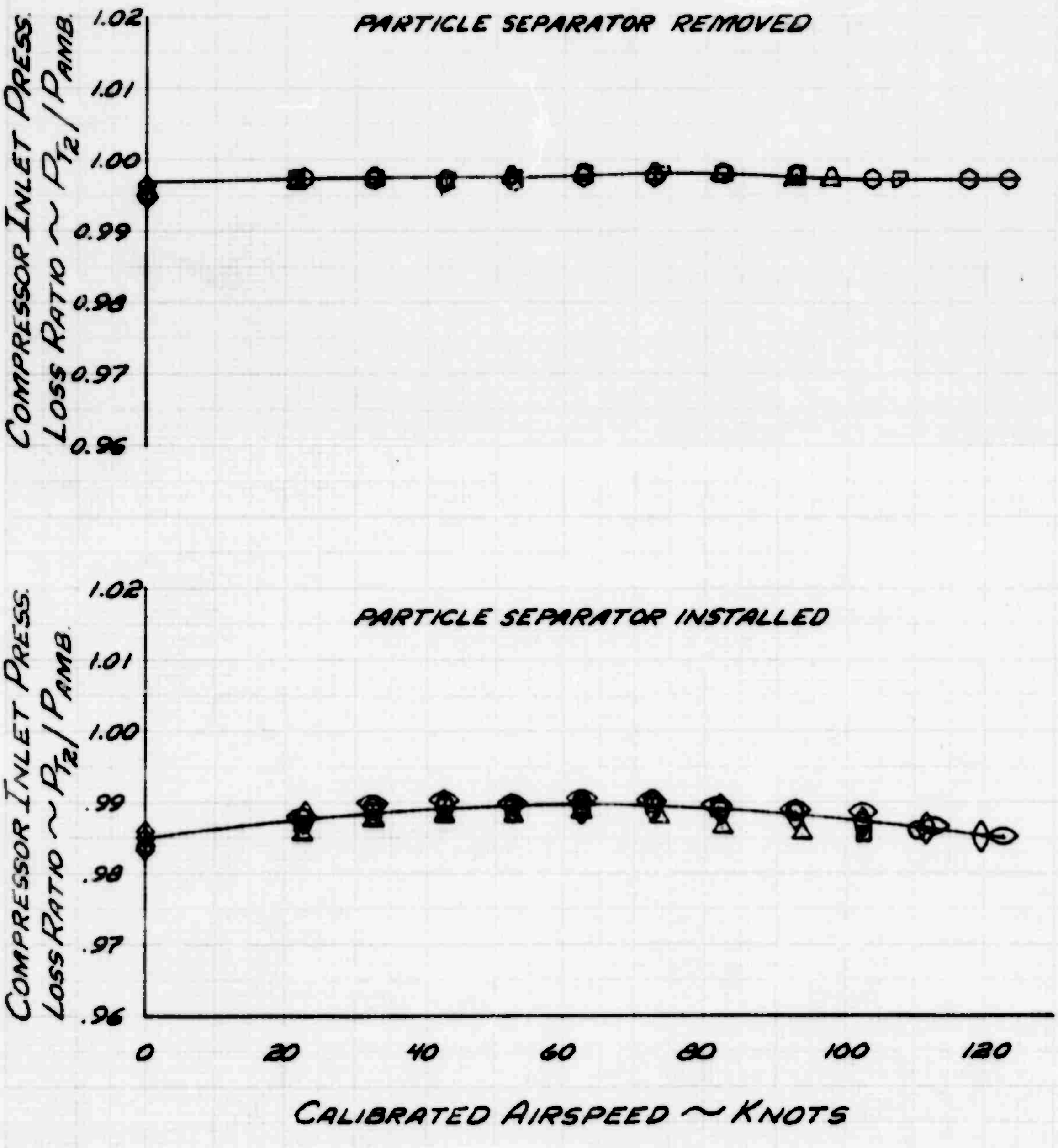


FIGURE 61

EFFECT OF BELL EXHAUST EXTENSION
ON T63-A-700 EXHAUST STATIC PRESSURE
OH-58A USA S/N 68-16693

NOTE: 1. DATA FURNISHED BY BELL HELICOPTER COMPANY

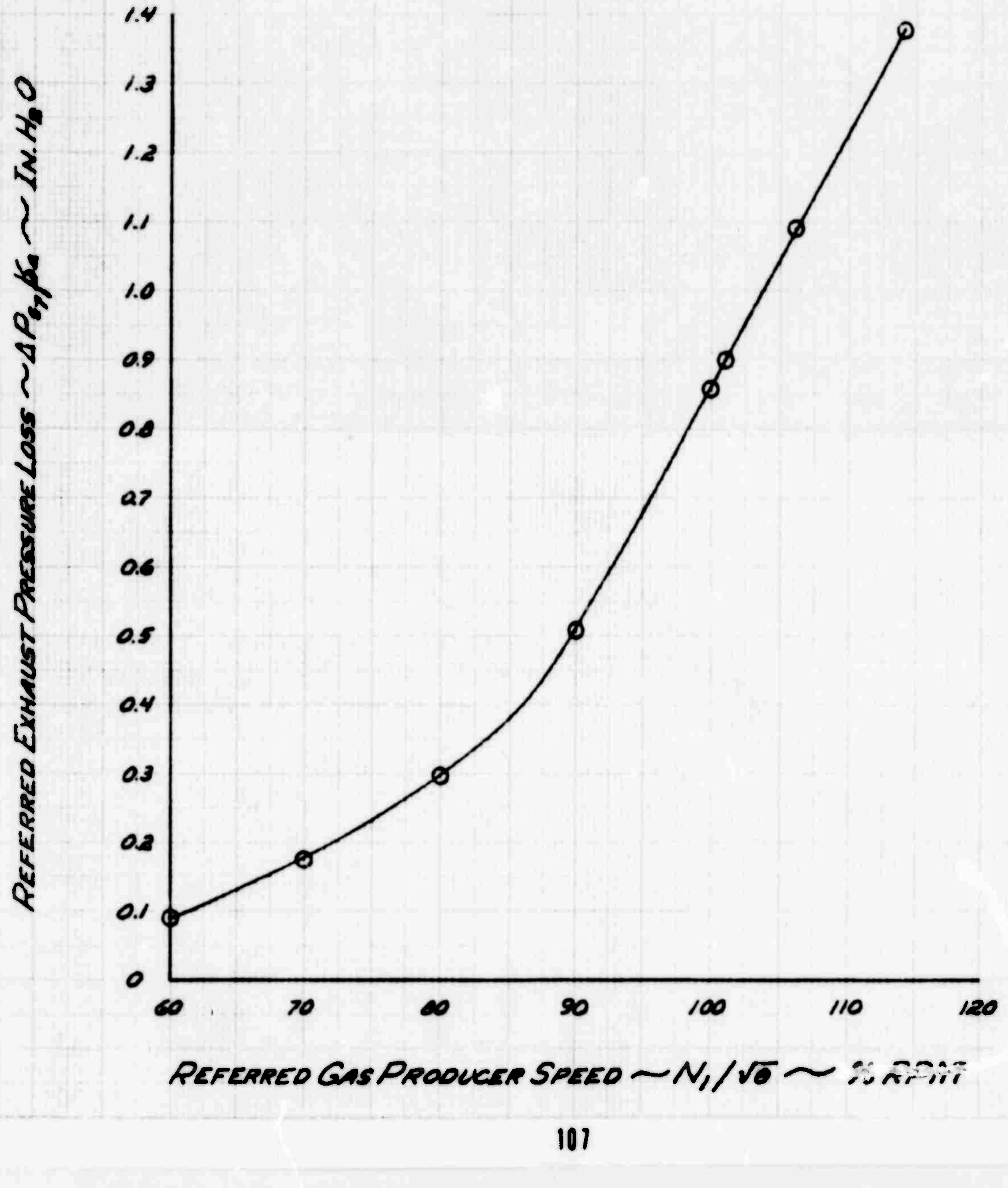


FIGURE 62
ENGINE CHARACTERISTICS
OH-58A USAS/N 68-16693
T63-A-700 S/N402811

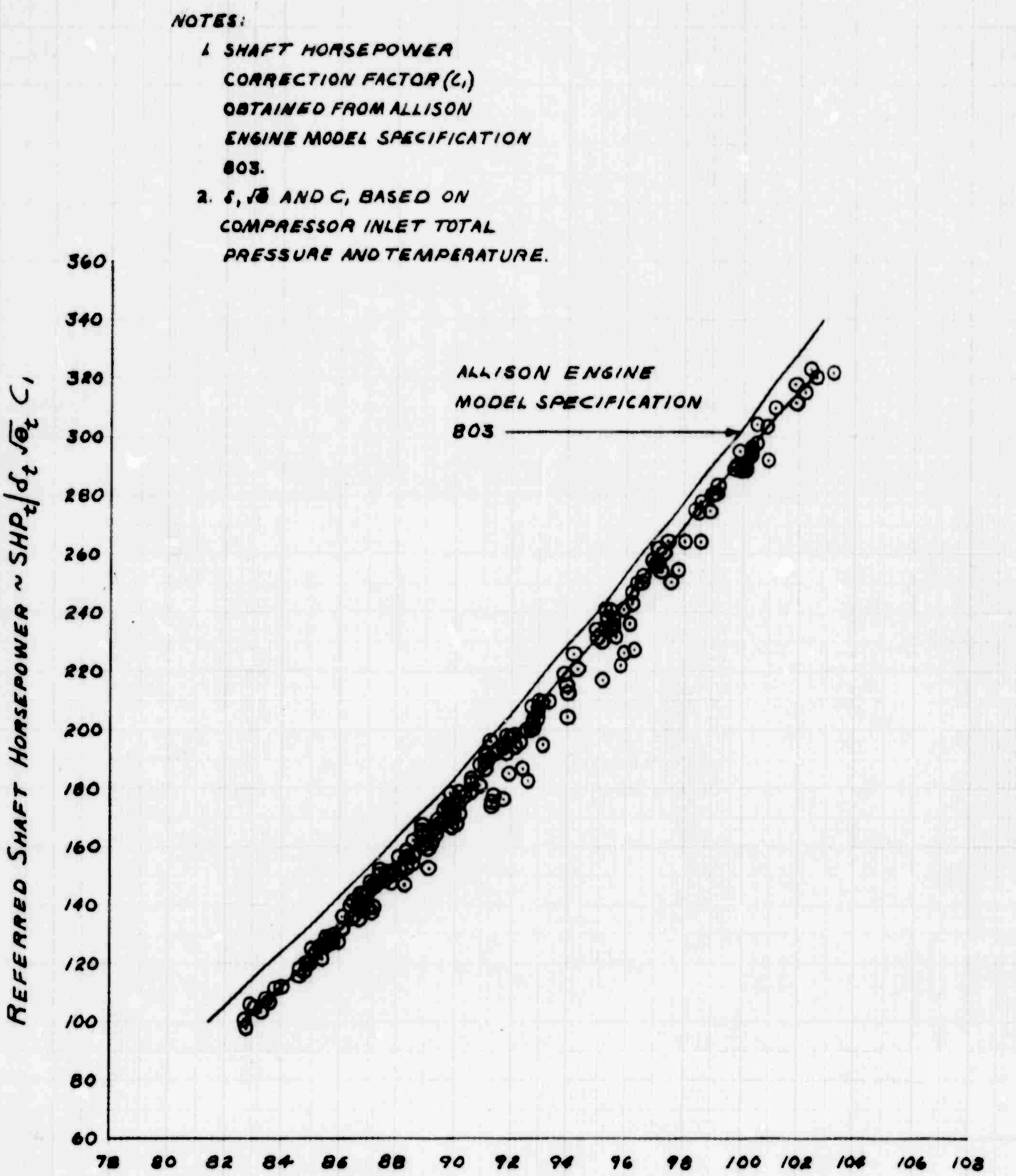


FIGURE 63
ENGINE CHARACTERISTICS
OH-58A USA S/N 68-16693
T63-A-700 S/N 402811

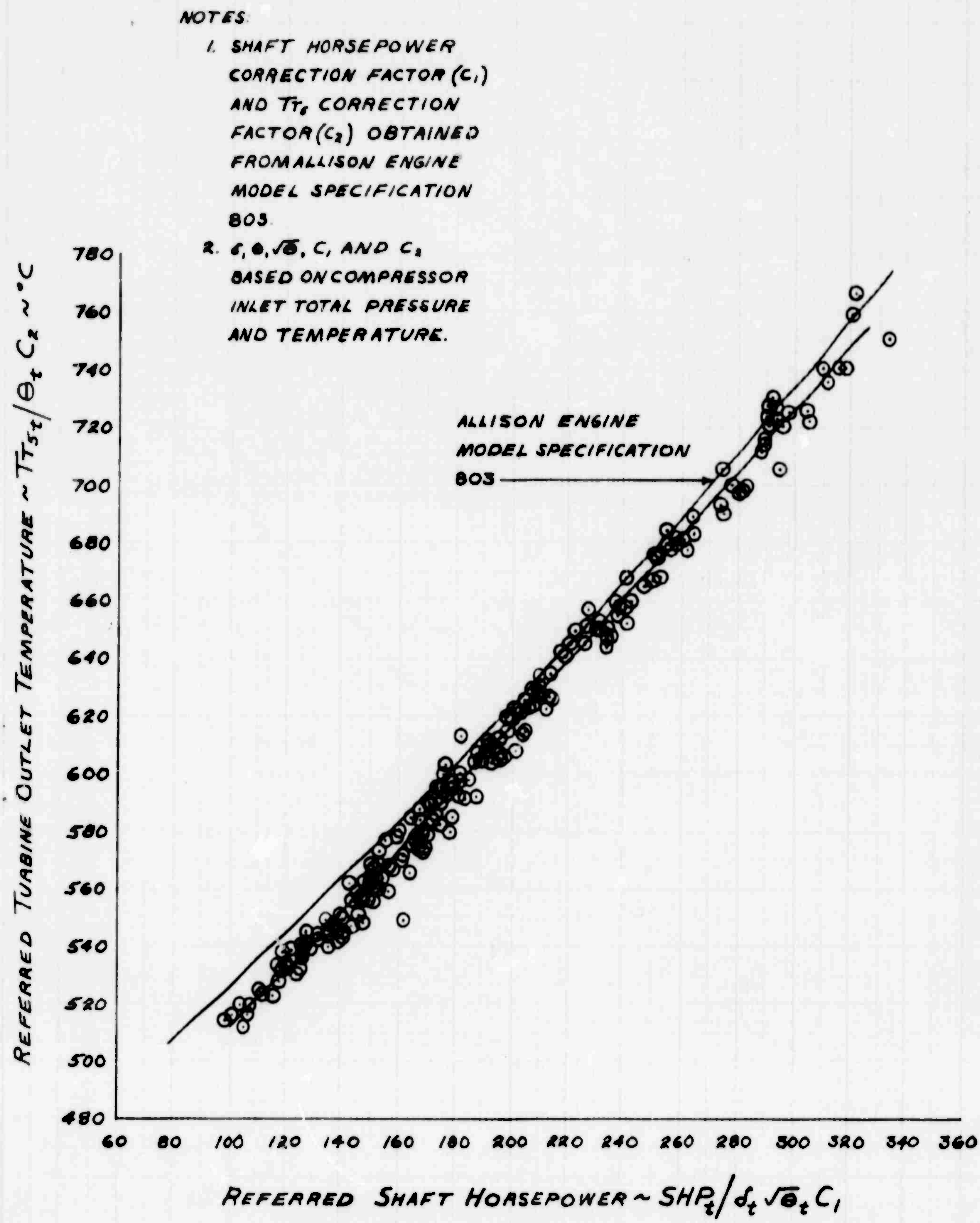


FIGURE 64
AIRSPEED CALIBRATION
OH-58A USA S/N 68-16693
SHIPS SYSTEM IN LEVEL FLIGHT

SYM	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)	CG STATION (IN)	DOOR CONF	CAL METHOD
○	3720	2548	354	106.8 (FWD)	ON	GROUND COURSE
□	3720	2548	347	106.8 (FWD)	ON	GROUND COURSE
△	6345	2587	354	107.6 (FWD)	OFF	TRAILING BOMB

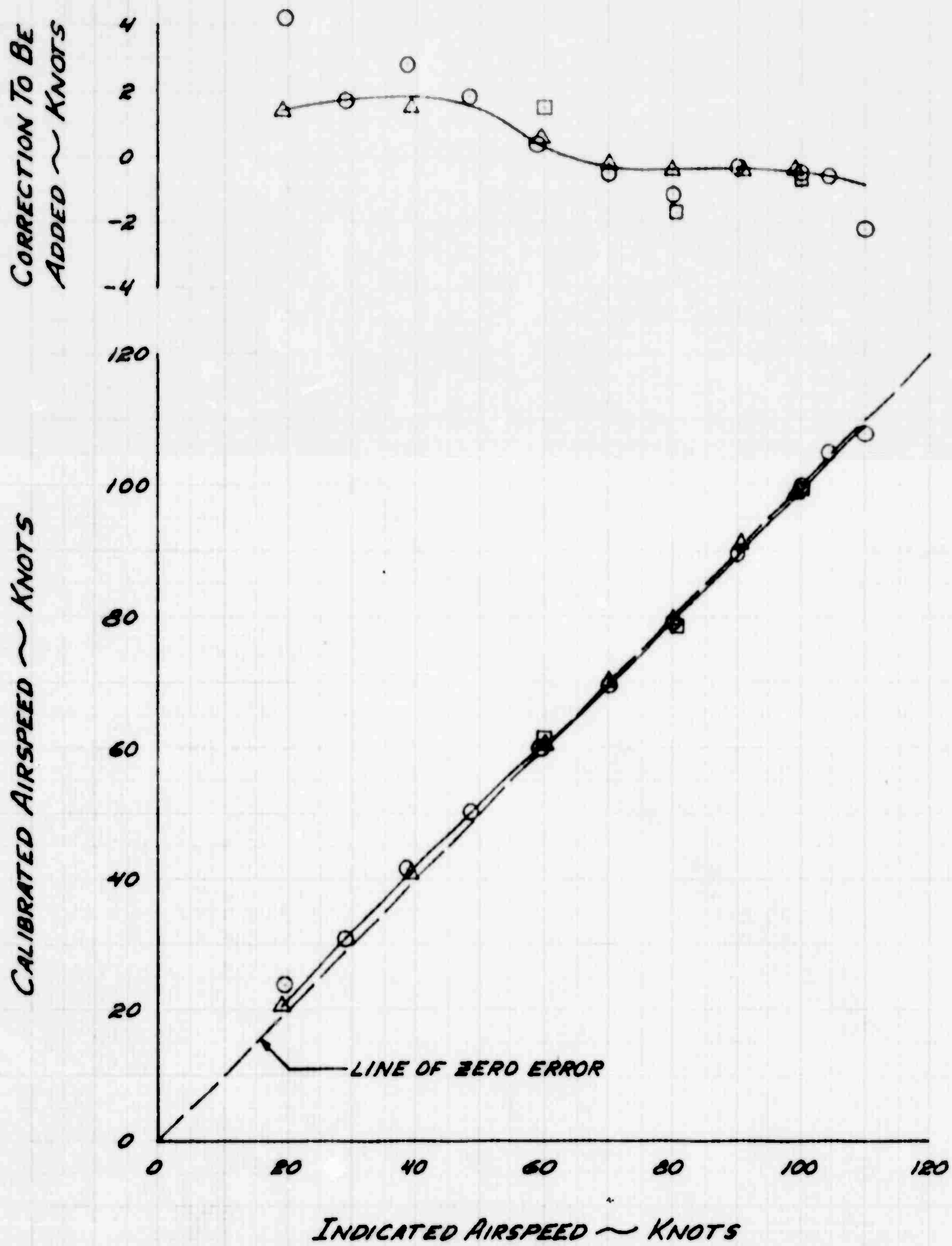


FIGURE 65
AIRSPEED CALIBRATION
OH-58A USA S/N 68-16693
SHIPS SYSTEM IN CLIMB
MILITARY RATED POWER

SYM	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)	CG STATION (IN)	DOOR CONF.	CAL METHOD
○	7230	2525	354	106.7 (FWD)	ON	TRAILING BOMB
□	6850	2587	354	107.6 (FWD)	OFF	TRAILING BOMB

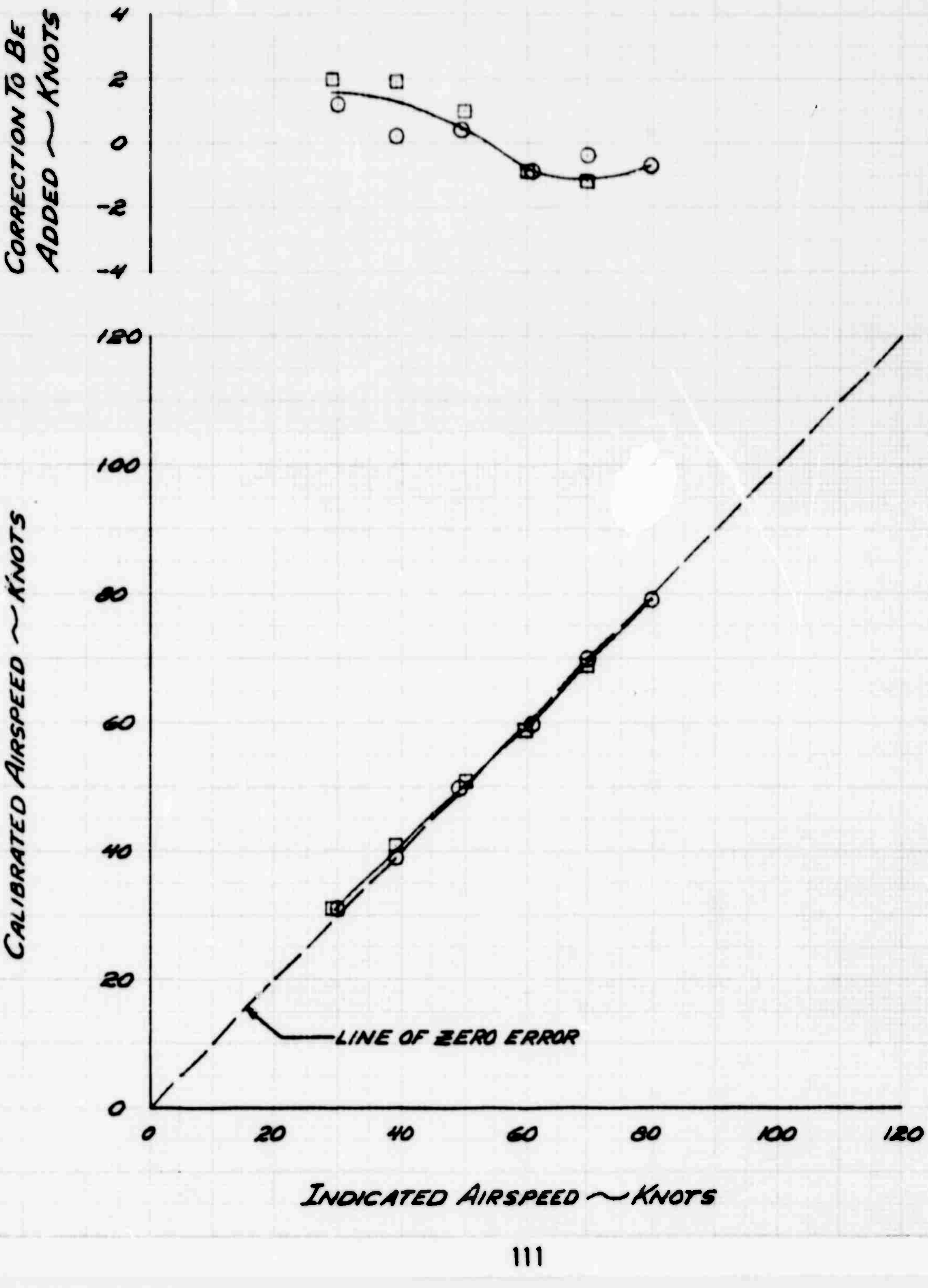


FIGURE 66
AIRSPEED CALIBRATION
OH-58A USA S/N 68-16693
SHIPS SYSTEM IN AUTOROTATION

SYM	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)	CG STATION (IN)	DOOR CONF	CAL METHOD
○	7355	2525	354	106.7 (FWD)	ON	TRAILING BOMB
□	6510	2525	390	106.7 (FWD)	ON	TRAILING BOMB
△	7115	2525	330	106.7 (FWD)	ON	TRAILING BOMB
▽	7065	2587	354	107.6 (FWD)	OFF	TRAILING BOMB

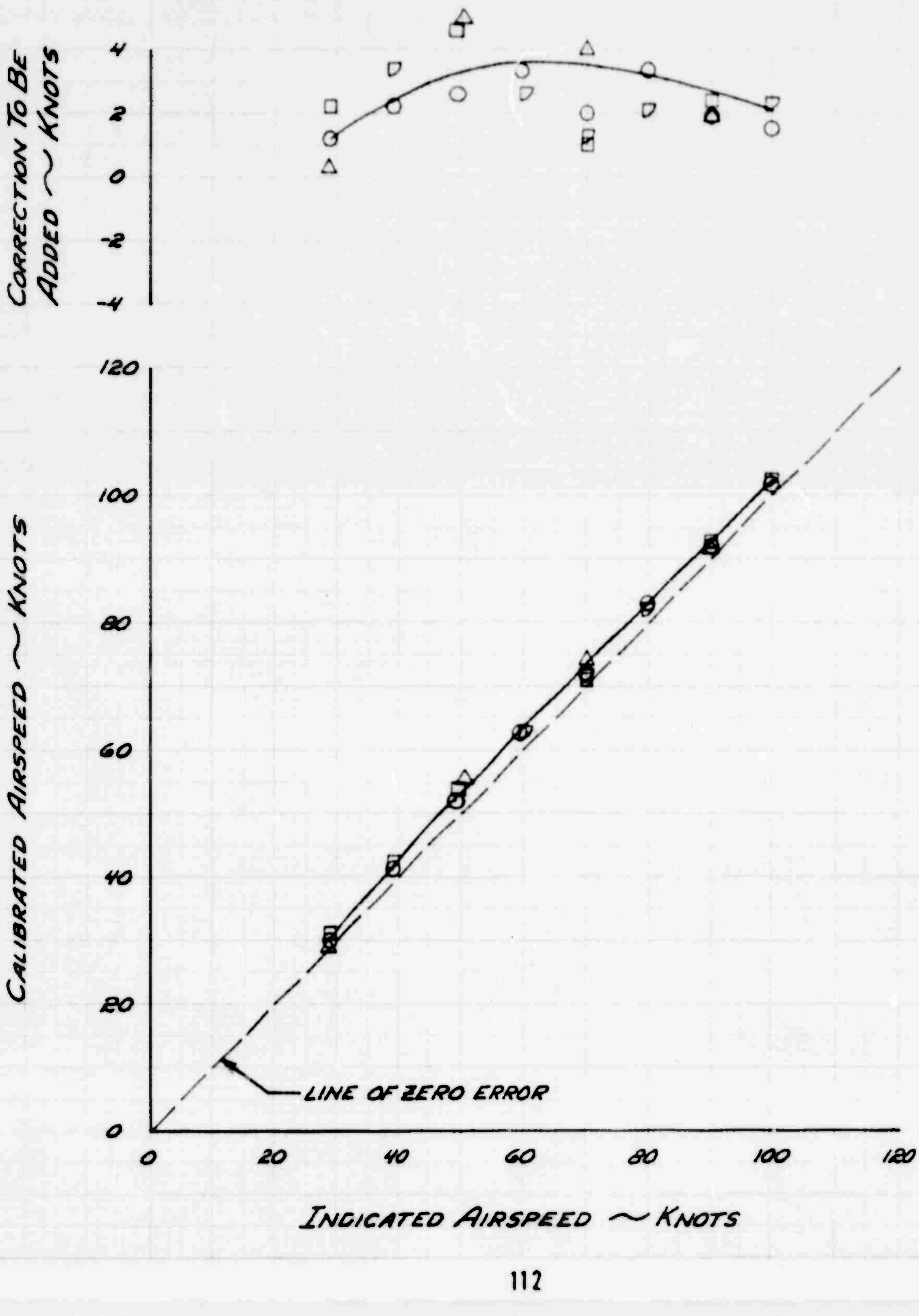


FIGURE 67

AIRSPEED CALIBRATION

OH-58A USA S/N 68-16693

TEST BOOM SYSTEM

CLEAN CONFIGURATION

TRAILING BOMB METHOD

SYM	FLIGHT COND	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)	CG STATION (IN)
○	LF	5890	2490	354	107.6 (FWD)
□	LF	5750	2485	347	107.5 (FWD)
◇	C	6720	2460	354	107.4 (FWD)
△	AUTO	7515	2440	354	107.3 (FWD)
◁	AUTO	7735	2385	330	106.9 (FWD)
▽	AUTO	7270	2345	390	106.8 (FWD)

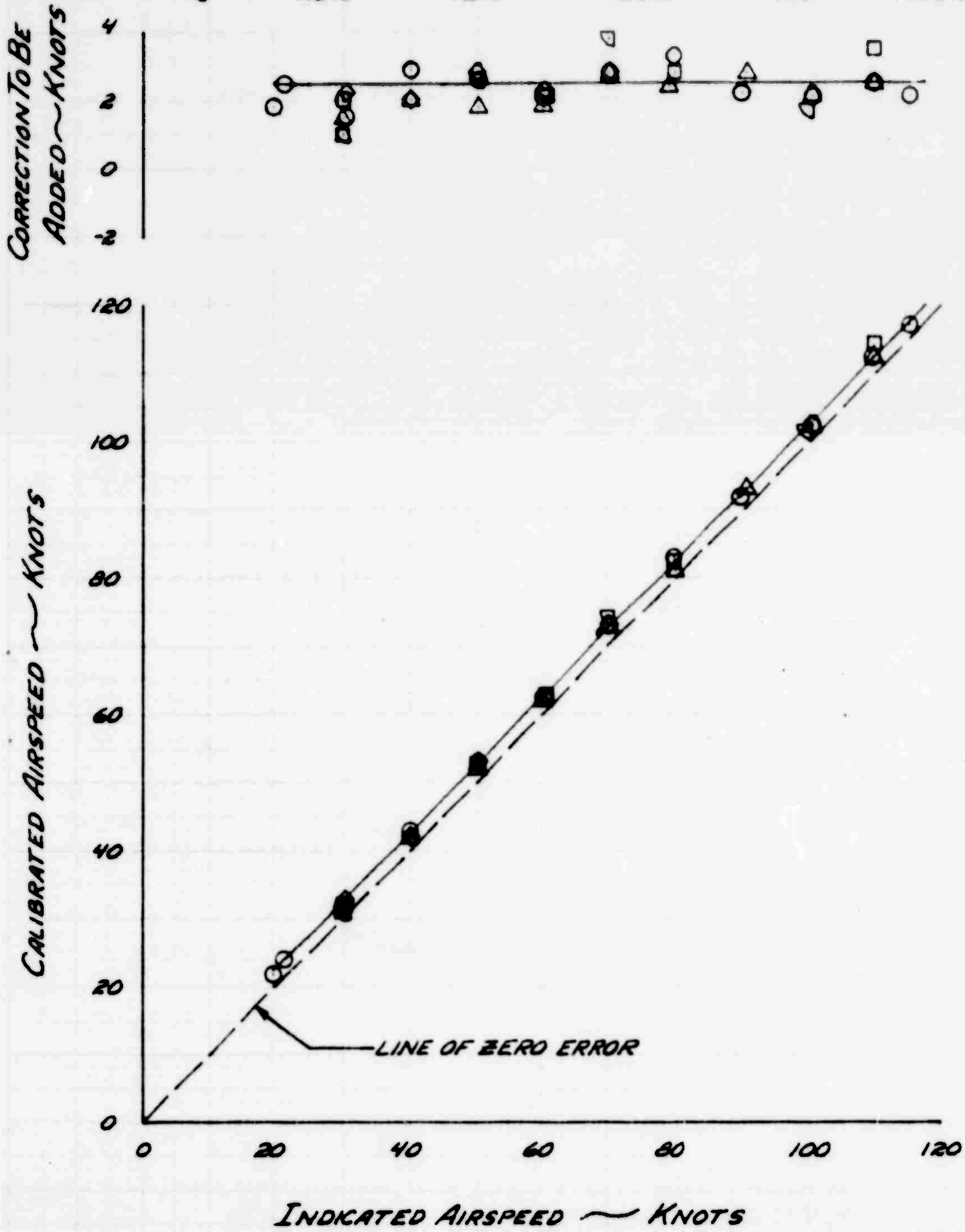


FIGURE 68
ALTIMETER CALIBRATION FOR STANDARD SYSTEM
OH-58A USA S/N 68-16693

SYM	PRESSURE ALTITUDE (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)	CG STATION (IN)	DOOR CONF	FLT CONDTN
○	2650	2548	354	106.8 (FWD)	ON	LEVEL
○	4810	2525	354	106.7 (FWD)	ON	CLIMB
○	5110	2525	354	106.7 (FWD)	ON	AUTO
♂	4795	2525	330	106.7 (FWD)	ON	AUTO
△	2550	2548	347	106.8 (FWD)	ON	LEVEL
△	4415	2525	390	106.7 (FWD)	ON	AUTO
□	3830	2587	354	107.6 (FWD)	OFF	LEVEL
□	4370	2587	354	107.6 (FWD)	OFF	CLIMB
□	4775	2587	354	107.6 (FWD)	OFF	AUTO

NOTE: ALTIMETER POSITION ERROR DERIVED FROM AIRSPEED POSITION ERROR DATA, FIGS 64 THRU 66

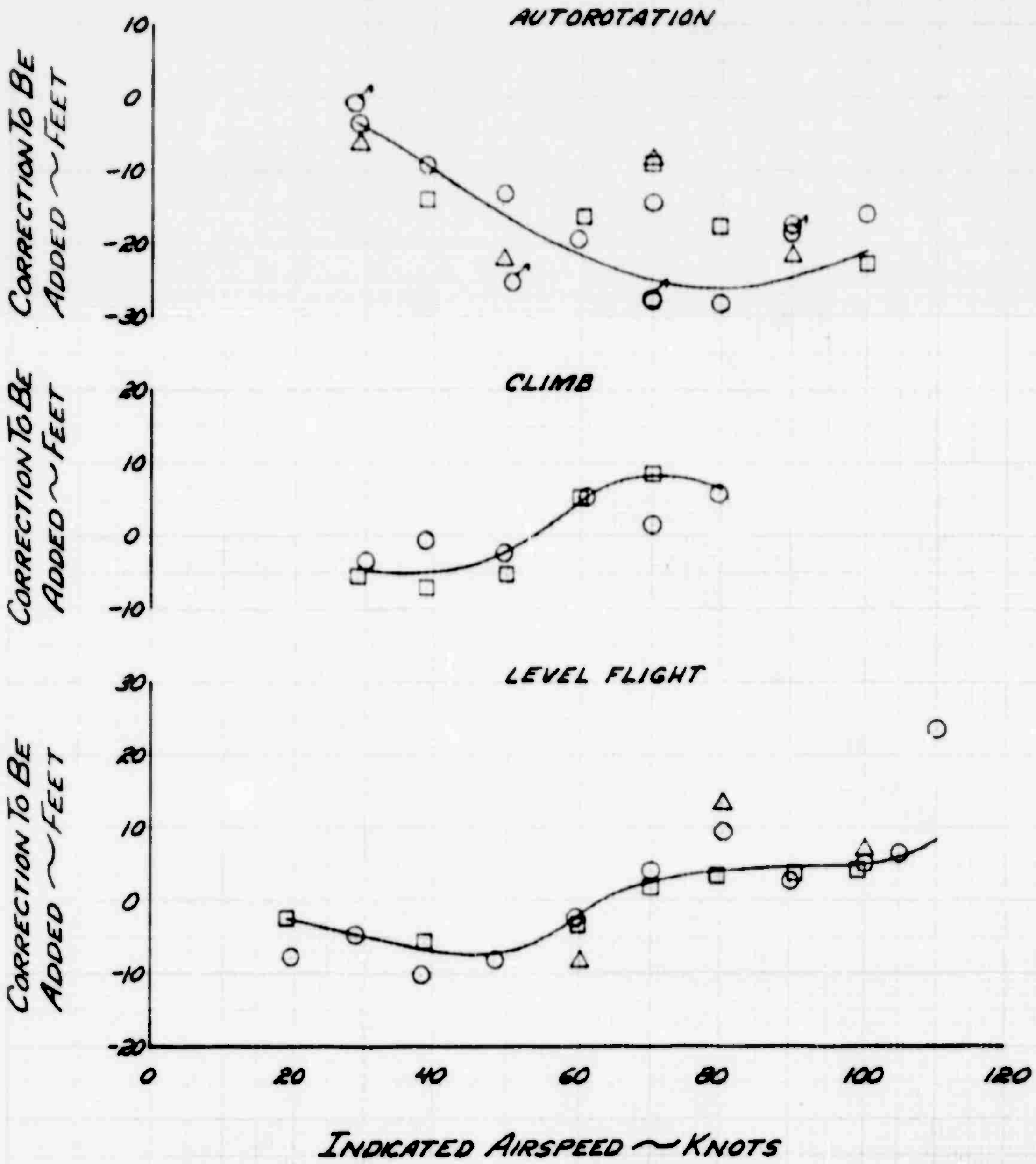


FIGURE 69
SHAFT HORSEPOWER AVAILABLE COMPARISON
OH-58A USA S/N 68-16693
T63-A-700 ENGINE
TAKEOFF RATED POWER T_{T_5} = 1380°F

NOTE: 1. STATIC CONDITIONS.
2. 0 BLEED AND ANTI-ICE OFF.
3. POWER EXTRACTED EQUALS 2.0 SHP.
4. INLET LOSSES BASED ON FIGS. 59 AND 60
5. EXHAUST PRESSURE LOSS IS 1.16 INCHES OF H_2O, FIG. 61
6. SHAFT HORSEPOWER DERIVED FROM ENGINE MODEL SPECIFICATION 803 T63-A-700 17 JUL 68

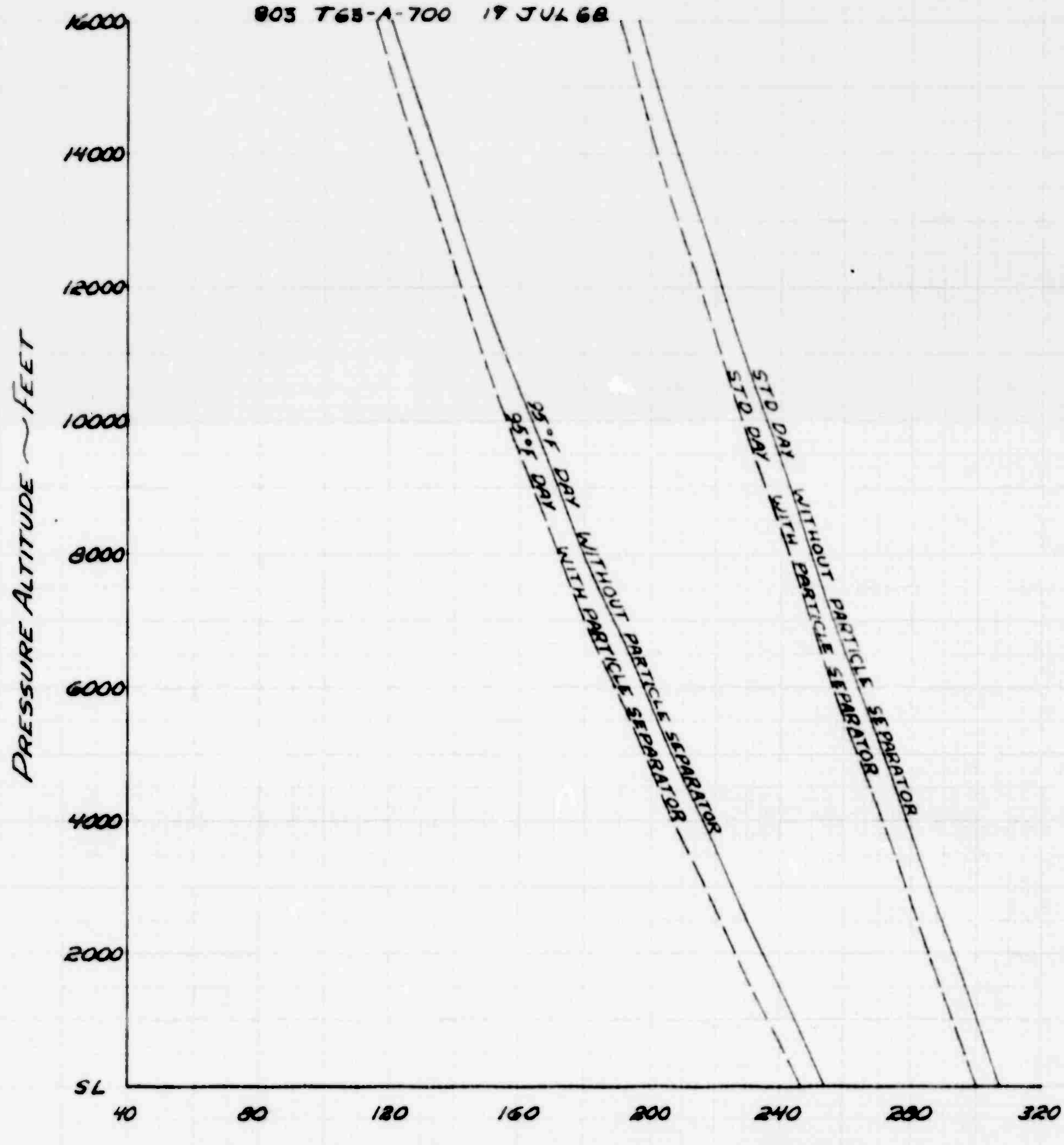


FIGURE 70

SHAFT HORSEPOWER AVAILABLE

OH-58A USA S/N 68-16693

T63-A-700 ENGINE

TAKEOFF RATED POWER T_{T_5} = 1380 °F

PARTICLE SEPARATOR INSTALLED 354 RPM

NOTE:
1. STATIC CONDITIONS.
2. 0 BLEED AND ANTI-ICE OFF.
3. POWER EXTRACTED EQUALS 2.0 SHP.
4. INLET LOSSES BASED ON FIGS. 59 AND 60
5. EXHAUST PRESSURE LOSS IS 1.16 INCHES OF H_2O, FIG 61
6. SHAFT HORSEPOWER DERIVED FROM ENGINE MODEL SPECIFICATION 803 T63-A-700 19 JUL 68

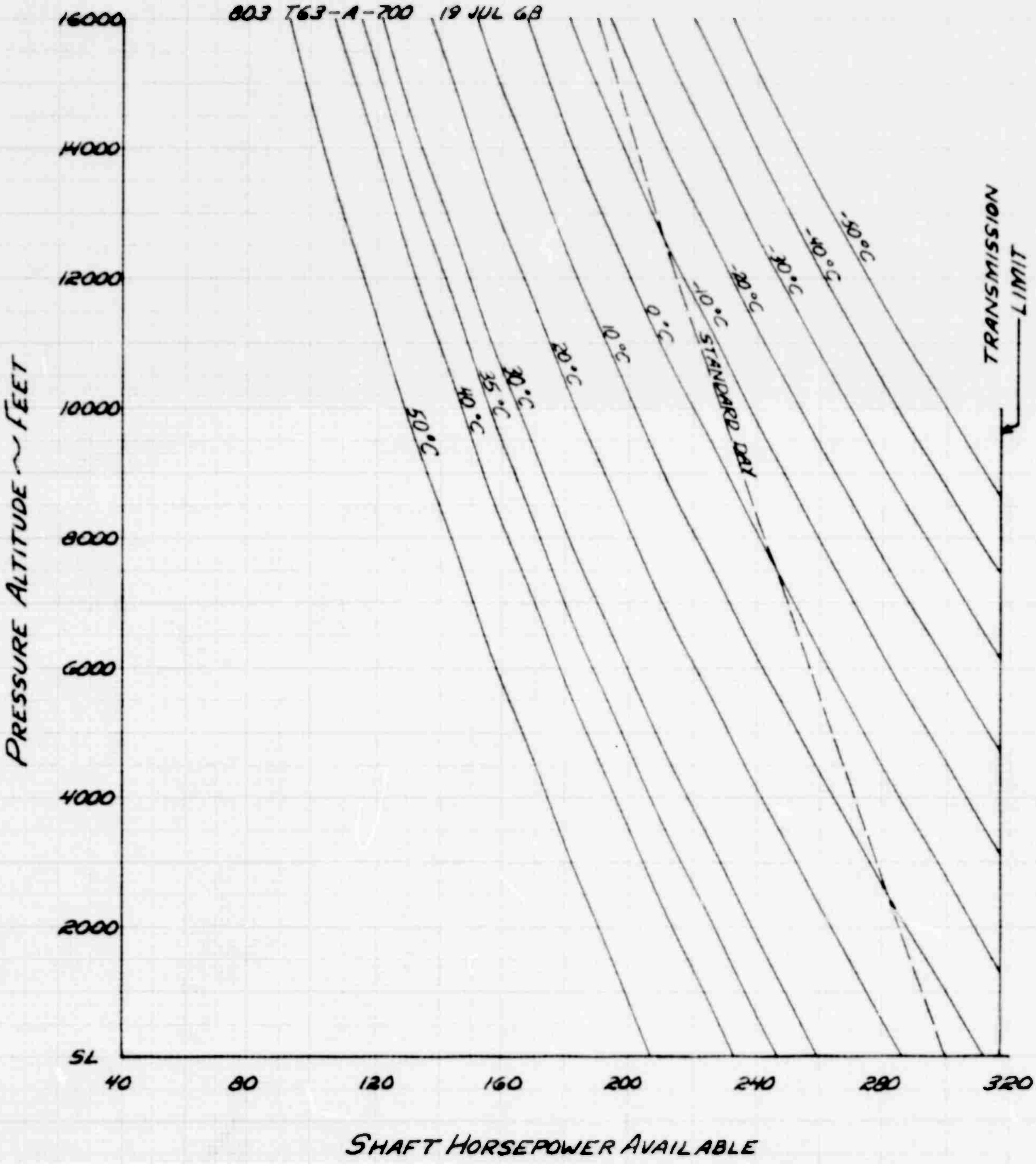


FIGURE 71

SHAFT HORSEPOWER AVAILABLE

OH-58A USA S/N 68-16693

T63-A-700 ENGINE

NORMAL RATED POWER T_{T_5} = 1280 °F

PARTICLE SEPARATOR INSTALLED 354 RPM

NOTE: 1. STATIC CONDITIONS.

2. 0 BLEED AND ANTI-ICE OFF.

3. POWER EXTRACTED EQUALS 2.0 SHP.

4. INLET LOSSES BASED ON FIGS. 59 AND 60

5. EXHAUST PRESSURE LOSS IS 1.16 INCHES OF H_2O, FIG. 61

6. SHAFT HORSEPOWER DERIVED FROM ENGINE MODEL SPECIFICATION 803 T63-A-700 19 JUL 68.

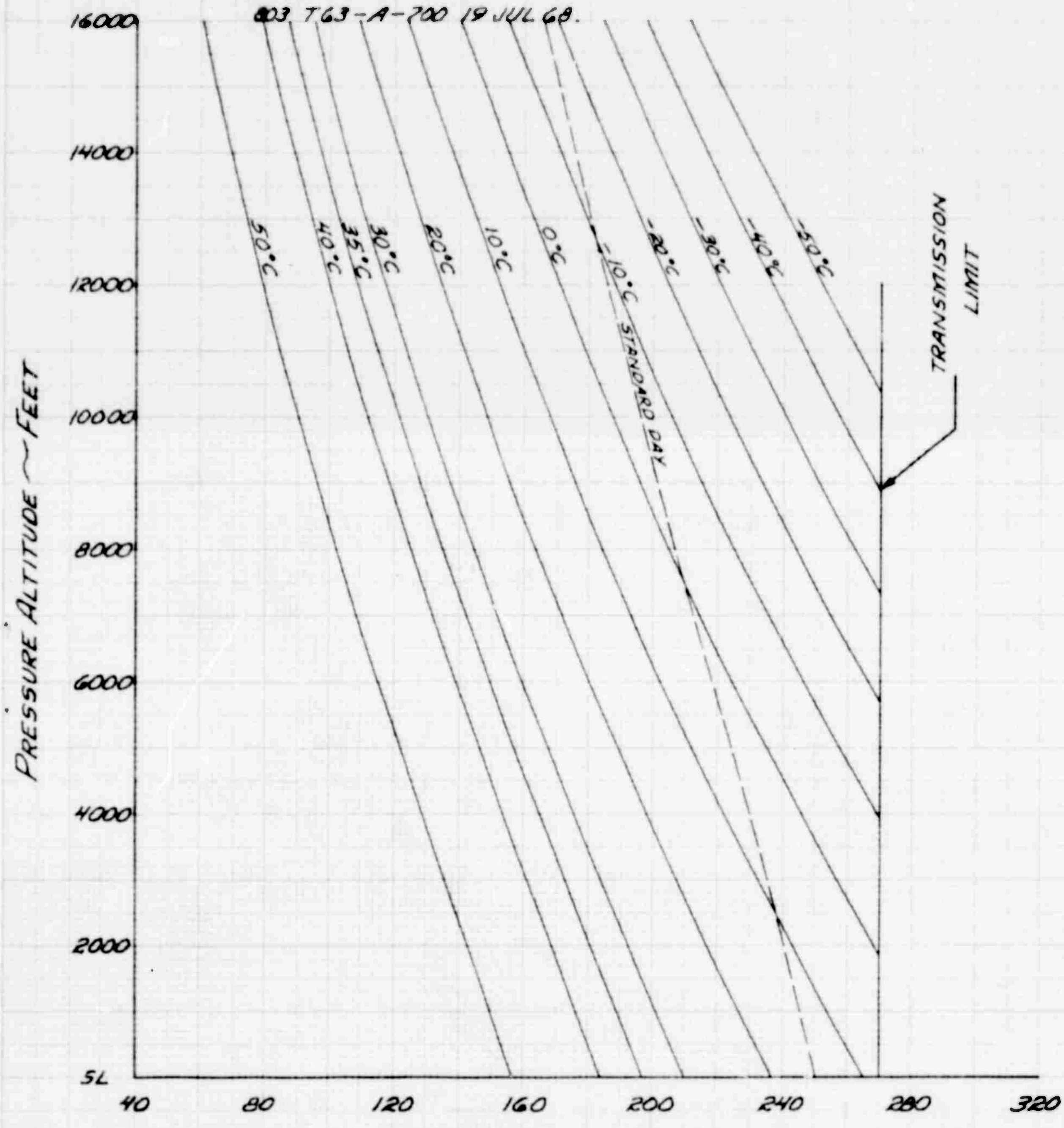
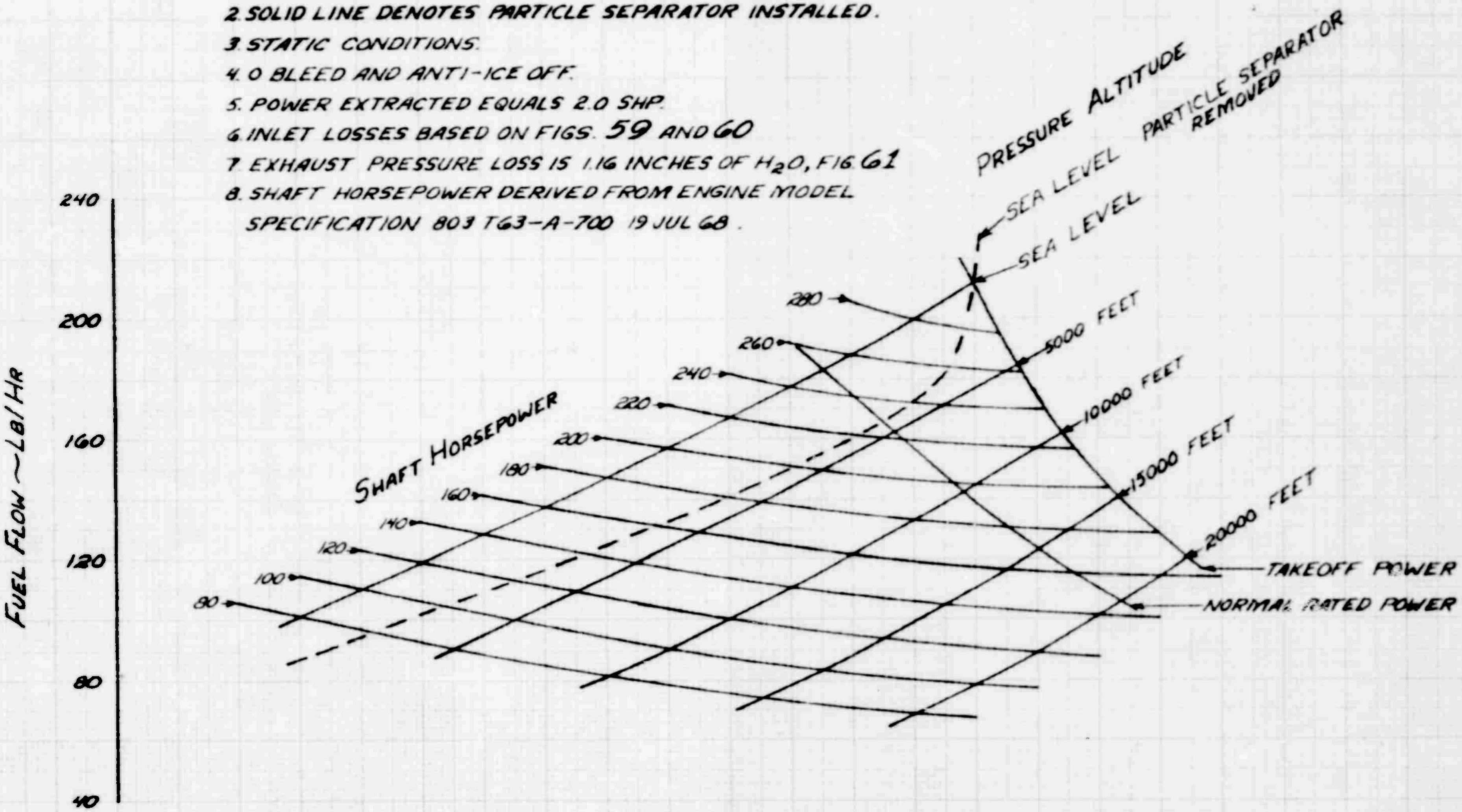


FIGURE 72
FUEL FLOW
T63-A-700 ENGINE
STANDARD DAY

NOTE: 1. FUEL FLOW BASED ON ALLISON MODEL SPECIFICATION NO. 803, MODEL T63-A-700, 19 JUL 68.
2. SOLID LINE DENOTES PARTICLE SEPARATOR INSTALLED.
3. STATIC CONDITIONS.
4. 0 BLEED AND ANTI-ICE OFF.
5. POWER EXTRACTED EQUALS 2.0 SHP.
6. INLET LOSSES BASED ON FIGS. 59 AND 60
7. EXHAUST PRESSURE LOSS IS 1.16 INCHES OF H_2O, FIG. 61
8. SHAFT HORSEPOWER DERIVED FROM ENGINE MODEL SPECIFICATION 803 T63-A-700 19 JUL 68.



APPENDIX VII. SYMBOLS AND ABBREVIATIONS

SYMBOL	DEFINITION
δ_a	Pressure ratio based on ambient condition
δ_t	Pressure ratio based on test condition
Δ	Difference in value
ΔCIP_{ic}	Indicated compressor inlet pressure difference $(P_{T_2} - P_a)$ corrected for instrument error (in. of Hg)
ΔC_P	Excess power between installed test aircraft power available minus horsepower required to hover at a reference skid height; nondimensional factor for excess power
ΔH_{PC}	Altimeter position error
$\Delta R/C_p$	Difference in rate of climb due to power differences
$\Delta R/C_w$	Difference in rate of climb due to weight
ΔSHP	Difference in standard shaft horsepower available and test shaft horsepower measured
ΔT	Temperature difference (°C)
ΔV_{PC}	Airspeed position error
μ	Airspeed ratio; nondimensional unit of airspeed
ρ	Density; air density
ρ_s	Standard-day density
ρ_t	Test-day density
σ	Density ratio

SYMBOL	DEFINITION
σ_s	Standard-day air density ratio at the test indicated altitude
$\sqrt{\sigma_{avg}}$	Average density ratio at the average density altitude
θ_t	Temperature ratio based on test condition
A	Main rotor disc area
a_{SL}	661.48 knots
C_1	Shaft horsepower correction factor
C_2	Gas producer and turbine outlet total temperature correction factor
C_3	Fuel-flow correction factor
C_P	Power coefficient; nondimensional unit of power; coefficient of power
C_{P_s}	Power coefficient based on standard condition
C_{P_t}	Power coefficient based on test condition
CIT; T_{T_2}	Compressor inlet total temperature
CIT_{ic}	Indicated compressor inlet total temperature corrected for instrument error (°C)
cps	Cycles per second
C_T	Thrust coefficient; nondimensional unit of thrust or weight; coefficient of thrust
C_{T_s}	Thrust coefficient based on standard condition

SYMBOL	DEFINITION
C_{T_t}	Thrust coefficient based on test condition
°C	Degree(s) centigrade
°F	Degree(s) Fahrenheit
°K	Degree(s) Kelvin
$\frac{dhp}{dt}$	Slope of pressure altitude versus time curve at a given pressure altitude
F_e	Equivalent flat plate area
FU	Fuel used
GR	Gear ratio of the output shaft rotational speed to the main rotor rotational speed
GW	Gross weight
GW_s	Standard gross weight
GW_t	Test gross weight
H_D	Density altitude
H_p	Pressure altitude
Hz	Hertz
in. of Hg	Inches of mercury
in. of H_2O	Inches of water
K_p	Power correction factor; coefficient of power correction; correction factor for variation in power
K_t	Conversion factor to change measured engine torque pressure (psi) to ft-lb
K_w	Weight correction factor; coefficient of weight correction; correction factor for variation in gross weight

SYMBOL	DEFINITION
$M_{N_{tip}}$	Advancing tip Mach; advancing tip Mach number
N_1	Gas producer speed
N_{1_t} ; $\sqrt{\theta_t}$	Referred gas producer speed at test condition
N_2	Power turbine speed
N_R	Main rotor speed
P_a	Ambient total pressure
P_{a_c}	P_a at the indicated pressure altitude plus ΔP_a for altitude position error using formula 7
P_{S_7}	Exhaust pressure
P_{T_2}	Compressor inlet total pressure
R	Radius; main rotor radius
R/C	Rate of climb
R/C_s	Standard rate of climb; final rate of climb standard
R/C_T	Tapeline rate of climb
R/D	Rate of descent
R/D_T	Tapeline rate of descent
S	Course length
SHP_{avail}	Shaft horsepower available for the installed test engine at takeoff atmospheric conditions
SHP_{req} at 2 ft	Shaft horsepower required for a 2-foot hover skid height at takeoff atmospheric conditions

SYMBOL	DEFINITION
SHP_s	Standard shaft horsepower obtained from a model specification engine
SHP_t	Test shaft horsepower obtained from a model specification engine
S/N	Serial number
S/S	Sideslip angle
t	Time required to travel the course distance
$T_{a_{ic}}$	Indicated ambient temperature corrected for instrument error
TOP	Takeoff power
TOT; T_{t_2}	Turbine outlet temperature
TOT; T_{T_5}	Turbine outlet total temperature
TRQ	Torque
T_t	Test ambient air temperature at the pressure altitude at which the slope is taken
$T_{T_{5_t}}$; θ_t	Referred turbine outlet total temperature
T_s	Standard ambient air temperature at the pressure altitude at which the slope is taken
V_{cal}	Calibrated airspeed
$V_{cal_{bomb}}$	Bomb system indicated airspeed corrected for instrument error
$V_{cal_{std}}$	Standard calibrated airspeed obtained from the ground speed course
V_{IC}	Test system indicated airspeed corrected for instrument error

SYMBOL	DEFINITION
$V_{IC_{test}}$	Test system indicated airspeed corrected for instrument error
V_{NE}	Never exceed airspeed
V_T	True airspeed
W	Gross weight; aircraft gross weight
W_f	Fuel flow; fuel-flow rate
W_s	Standard-day gross weight
W_t	Test-day gross weight
W/ρ	Gross weight divided by density

ABBREVIATION	DEFINITION
AC	Alternating current
AVG, avg	Average
BHC	Bell Helicopter Company
CG, cg	Center of gravity
DC	Direct current
deg	Degree/degrees
FAA	Federal Aviation Administration
fig., figs.	Figure, figures
fpm	Foot/feet per minute
FS	Fuselage station
ft	Foot/feet
fwd	Forward
GRWT, grwt	Gross weight

ABBREVIATION	DEFINITION
hr, hrs	Hour, hours
in.	Inch/inches
KCAS	Knots calibrated airspeed
KTAS	Knots true airspeed
lb	Pound/pounds
LOH	Light observation helicopter
MAX, max	Maximum
MIN, min	Minimum
MRP	Military rated power
NAMPP	Nautical air miles per pound of fuel
NAMT	Nautical air miles traveled
NRP	Normal rated power
OAT	Outside air temperature
OGE	Out of ground effect
rpm	Revolutions per minute
SHP, shp	Shaft horsepower
SL	Sea level

SUBSCRIPT	DEFINITION
a	Ambient
s	Standard; standard day
std	Standard
t	Test; test day
T	Tapeline
1	Reciprocal heading
2	Reciprocal heading

APPENDIX VIII. DISTRIBUTION

Agency	Test Plans	Interim Reports	Final Reports
Commanding General US Army Aviation Systems Command			
ATTN: AMSAV-R-F	5	5	6
AMSAV-C-A	-	-	2
AMSAV-D-ZDOR	-	-	2
AMSAV-D-W	-	-	2
AMSAV-R-R	-	-	1
PO Box 209 St. Louis, Missouri 63166			
Commanding General US Army Materiel Command			
ATTN: AMCPM-LH	5	1	5
PO Box 209 St. Louis, Missouri 63166			
Commanding General US Army Materiel Command			
ATTN: AMCRD	2	1	2
AMCAD-S	-	-	1
AMCPM-AI	5	1	5
AMCPP	-	-	1
AMCMR	2	-	2
AMCQA	-	-	1
Washington, D. C. 20315			
Commanding General US Army Combat Developments Command			
ATTN: USACDC LnO	11	11	11
PO Box 209 St. Louis, Missouri 63166			
Commanding General US Continental Army Command			
ATTN: DCSIT-SCH-PD	-	-	1
Fort Monroe, Virginia 23351			

Agency	Test Plans	Interim Reports	Final Reports
Commanding General US Army Test and Evaluation Command ATTN: AMSTE-BG	2	2	2
USMC LnO	1	1	1
Aberdeen Proving Ground, Maryland 21005			
Commanding Officer US Army Aviation Materiel Laboratories ATTN: SAVFE-SO, M. Lee	-	-	1
SAVFE-TD	-	-	2
SAVFE-AM	-	-	1
SAVFE-AV	-	-	1
SAVFE-PP	-	-	1
Fort Eustis, Virginia 23604			
Commanding General US Army Aviation Center Fort Rucker, Alabama 36362	1	1	1
Commandant US Army Primary Helicopter School Fort Wolters, Texas 76067	1	1	1
President US Army Aviation Test Board Fort Rucker, Alabama 36362	1	1	1
Director US Army Board for Aviation Accident Research Fort Rucker, Alabama 36362	-	1	1
President US Army Maintenance Board Fort Knox, Kentucky 40121	-	-	1
Commanding General US Army Electronics Command ATTN: AMSEL-VL-D Fort Monmouth, New Jersey 07703	-	-	1

Agency	Test Plans	Interim Reports	Final Reports
Commanding General US Army Weapons Command ATTN: AMSWE-RDT AMSWE-REW (Airborne Armament Flying) Rock Island Arsenal Rock Island, Illinois 61202	- -	- -	2 2
Commandant US Marine Corps Washington, I C. 20315	-	-	1
Director US Marine Corps Landing Force Development Center Quantico, Virginia 22133	1	1	2
US Air Force Aeronautical Systems Division ATTN: ASNFD-10 Wright Patterson Air Force Base, Ohio 45433	-	-	1
Air Force Flight Test Center ATTN: PSD SYSE Edwards Air Force Base, California 93523	- -	- -	5 2
Naval Air System Command Headquarters (A530122) Department of the Navy Washington, D. C. 20350	-	-	1
Commander Naval Air Test Center (FT23) Patuxent River, Maryland 20670	1	-	1
Federal Aviation Administration ATTN: Administrative Standards Division (MS-110) 800 Independence Avenue S.W. Washington, D. C. 20590	-	-	2

Agency	Test Plans	Interim Reports	Final Reports
Department of the Army Office of the Chief, Research and Development ATTN: CRD Washington, D. C. 20310	7	-	7
Department of the Army Deputy Chief of Staff for Logistics ATTN: LOG/MED	-	-	1
LOG/SAA-ASLSB Washington, D. C. 20310	-	-	1
Director US Army Aeromedical Research Unit Fort Rucker, Alabama 36362	-	-	1
Bell Helicopter Company Military Marketing Sales Engineering PO Box 482 Fort Worth, Texas 79901	-	-	5
Allison Division of General Motors Corporation PO Box 894 Indianapolis, Indiana 46206	-	-	5
Defense Documentation Center Cameron Station Alexandria, Virginia 22314	-	-	20

UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) US ARMY AVIATION SYSTEMS TEST ACTIVITY EDWARDS AIR FORCE BASE, CALIFORNIA 93523	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP

3. REPORT TITLE
AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST, PRODUCTION OH-58A HELICOPTER, UNARMED AND ARMED WITH XM27E1 WEAPON SYSTEM, PERFORMANCE

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)
Final Report, August 1968 through May 1970

5. AUTHOR(S) (First name, middle initial, last name)
GEORGE M. YAMAKAWA, Project Engineer
JOSEPH C. WATTS, Project Officer/Pilot

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
SEPTEMBER 1970	129	5

8a. CONTRACT OR GRANT NO. b. PROJECT NO. USAAVSCOM PROJECT NO. 68-30 c. USAASTA PROJECT NO. 68-30 d.	9a. ORIGINATOR'S REPORT NUMBER(S) USAASTA PROJECT NO. 68-30 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) N/A

10. DISTRIBUTION STATEMENT
This document may be further distributed by any holder only with specific prior approval of the CG, USAAVSCOM, ATTN: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY US ARMY AVIATION SYSTEMS COMMAND ATTN: AMSAV-R-F, PO Box 209 PO Box 209, St. Louis, Missouri 63166

13. ABSTRACT

Performance tests were conducted on a production OH-58A helicopter to determine compliance with performance guarantees outlined in the detail specification and approved revisions, and also to provide information for the operator's manual. Testing was performed by the US Army Aviation Systems Test Activity, Edwards Air Force Base, California, during the period between 19 August 1969 and 13 January 1970. The testing consisted of 99 flights totaling 62.5 productive hours. The OH-58A met or exceeded all of the contractual performance guarantees. The engine inlet loses with the particle separator installed exceeded the limits of the detail specification. A single flight at cold temperatures indicated that level flight performance is significantly affected by compressibility. Although all contractual hover performance guarantees were met, the helicopter could not hover out of ground effect on a 95°F day at gross weights greater than 2930 pounds with the particle separator installed. Under moderate temperature conditions, the performance capabilities of the OH-58A are satisfactory for mission accomplishment. It is recommended that consideration be given to increasing the utility of the OH-58A by installing a more powerful engine.

DD FORM 1 NOV 65 1473

UNCLASSIFIED
Security Classification

UNCLASSIFIED
Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
OH-58A helicopter Performance test Particle separator						

UNCLASSIFIED
Security Classification