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AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST OF A SIXTH YEAR PRODUCTION UH - 60A

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FINAL REPORT

UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
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Testing was conducted to obtain performance data for inclusion in the sixth year production UH-60A helicopter operator's manual. A total of 58 productive flight hours were flown at five different test aitea between 3 and 20 October 1983 and 29 February and 18 September 1984. The out-of-ground effect hover gross weight capability was 16,526 pounds for 95 percent intermediate (30 minute limit) rated power available at 4700 feet preaaure altitude and 35°C temperature. The increase in equivalent flat plate area (F_e) of the sixth over

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year production aircraft configuration over the first year configuration was 5 square feet (ft^2) in level flight at a referred rotor speed ($N_R/\sqrt{\theta}$) of 258 revolutions per minute. Of this increase, 2.5 ft^2 was attributed to the External Stores Support System fixed provision fairings, 1.5 ft^2 to the external mounting brackets of the AN/ALQ-144(V) infrared countermeasures set and M-130 chaff dispenser, and 1.0 ft^2 to numerous other minor external changes. However, throughout the $N_R/\sqrt{\theta}$ range, the difference in power required between the first and sixth year production aircraft does not equate to a constant F_e. A limited investigation of the effect of stabilator position on level flight power required did not completely account for the power differences noted when flying at different dimensional conditions that produce the same nondimensional thrust coefficient.

N sub R/Square root theta

PREFACE

Special recognition is given to Vera L. Gardner for her innovative computer programming support which aided during the data analysis phase of this project.

TABLE OF CONTENTS

INTRODUCTION

BACKGROUND

1. The US Army has contracted with Sikorsky Aircraft Division of United Technologies, for the sixth, seventh and eighth year UH-60A production lots. The US Army Aviation Engineering Flight Activity (USAAEFA) has conducted testing on earlier YUH-60A and UH-60A helicopters to include the Government Competitive Tests, Preliminary Airworthiness Evaluations, Climatic Laboratory Tests, Artificial and Natural Icing Tests and an Airworthiness and Flight Characteristics (A&FC) evaluation. Further testing was needed to update previous test results of the first year production UH-60A for inclusion in the sixth year production UH-60A operator's manual.

2. In September 1983, USAAEFA was tasked by the US Army Aviation Systems Command (AVSCOM) (ref 1, app A) to conduct an A&FC evaluation of a UH-60A helicopter from the sixth year production lot.

TEST OBJECTIVES

3. The objectives of the A&FC evaluation were as follows:

a. To determine the performance change caused by the infrared countermeasures set AN/ALQ-144(V) and chaff dispenser M-130, and their external mounting brackets.

b. To obtain sufficient level flight performance data to update existing data for inclusion in the operator's manual.

c. To obtain sufficient hover performance data to update existing data for inclusion in the operator's manual.

DESCRIPTION

4. The UH-60A is a twin-turbine single-main rotor helicopter capable of transporting cargo, 11 combat troops, and weapons during day, night, visual meteorological conditions, and instrument meteorological conditions. The helicopter is powered by two General Electric T700-GE-700 turboshaft engines, each having an installed thermodynamic rating (30 minute limit) of 1553 shaft horsepower (SHP) (power turbine speed of 20,900 revolutions per minute (rpm)) at sea level, standard day static conditions. Installed dual engine power is transmission limited to 2828 SHP. The engines used during this evaluation were calibrated by the engine manufacturer. Two test aircraft were used during this evaluation: USA S/N 82-23748, a sixth year production Black

Hawk in the normal utility External Stores Support System (ESSS) fixed provision fairings configuration (defined as normal utility (ESSS) configuration) described in paragraph 5, and USA S/N 77-22716, a first year production aircraft incorporating airspeed and stabilator modifications similar to the sixth year aircraft in the normal utility configuration, the normal utility configuration with AN/ALQ-144(V) and M-130 external mounting brackets added, and the normal utility configuration with AN/ALQ-144(V) and M-130 sets installed.

5. Several modifications were incorporated to arrive at the normal utility (ESSS) configuration for the sixth, seventh and eighth year production lots. These include reorientation of the production airspeed pitot-static tubes, a modified stabilator schedule, and the addition of external mounting brackets for the AN/ALQ-144(V) infrared countermeasures set and M-130 chaff dispenser. Also included were the ESSS fixed provisions and fairings along with numerous other minor external configuration changes. A more detailed description of the UH-60A with descriptive photographs is available in appendix B, and additional information can be found in the Prime Item Development Specification (ref 2, app A) and in the operator's manual (ref 3).

TEST SCOPE

6. Hover and level flight performance tests were conducted at Edwards AFB (elevation 2302 feet), Bakersfield (488 feet), Bishop (4120 feet) and Coyote Flats (9980 feet), California and at Duluth (1430 feet), Minnesota. Sixth year production aircraft test flight hours totaled 74 of which 51 were productive. These tests were conducted between 29 February and 18 September 1984. Level flight performance tests were also conducted on a first year production aircraft between 3 and 20 October 1983 and totaled 12 hours of which 7 were productive. Flight restrictions and operating limitations observed throughout the evaluation are contained in the operator's manual (ref 3, app A) and in the airworthiness release issued by AVSCOM (ref 4). Testing was conducted in accordance with the test plan (ref 5) at the conditions shown in table 1.

Table 1. Test Conditions[1]

Type	Gross Weight (lb)	Longitudinal Center of Gravity (FS)	Density Altitude (ft)	Referred Rotor Speed (rpm)	Trim Airspeed (KTAS)
Hover	13240[2] to 22680	354.0	3300 to 10720	244 to 265	0
Level[3] Flight	14470 to 21690	347.4	3430 to 14000	245 to 286	41 to 170

NOTES:

[1]Tests were conducted at an approximate mid lateral center of gravity with the automatic flight control system on in the normal utility (ESSS) configuration, unless otherwise noted.
[2]Aircraft gross weight plus cable tension.
[3]Tests also conducted with the AN/ALQ-144(V) and M-130 mounting brackets added and with the complete AN/ALQ-144(V) and M-130 sets installed.

TEST METHODOLOGY

7. The flight test data were recorded by hand from test instrumentation displayed in the cockpit, by on-board magnetic tape recording equipment and via telemetry to the Real Time Data Acquisition and Processing System. A detailed listing of test instrumentation is contained in appendix C. Level flight performance tests were supplemented by test data from a first year production UH-60A adjusted for drag differences. Flight test techniques and data reduction procedures are described in appendix D.

RESULTS AND DISCUSSION

GENERAL

8. Testing was conducted to obtain performance data for inclusion in the UH-60A sixth year production helicopter operator's manual. At the hover performance guarantee conditions of 95 percent intermediate (30 minute limit) rated power available (IRP), 4700 feet pressure altitude (Hp) and 35°C, the out-of-ground effect (OGE) hover gross weight capability was 16,526 pounds. The difference in level flight power required does not equate to a constant equivalent flat plate area (F_e) between the first and sixth year production aircraft throughout the referred rotor speed ($N_R/\sqrt{\theta}$) range. The increase in F_e between the normal utility and normal utility (ESSS) configurations at $N_R/\sqrt{\theta}$ of 258 rpm was determined to be 5 square feet (ft^2). Of the total 5.0 ft^2, 2.5 ft^2 can be attributed to the ESSS fixed provision fairings and 1.5 ft^2 can be attributed to the external mounting brackets of the AN/ALQ-144(V) infrared countermeasures set and M-130 chaff dispenser. The remaining 1.0 ft^2 is attributable to numerous other minor external configuration changes. Installing the AN/ALQ-144(V) and M-130 sets increases F_e by an additional 0.5 ft^2. The effect of stabilator position does not completely account for the discrepancy in power required as a result of flying at dimensionally different conditions that produce the same nondimensional thrust coefficient (C_T).

HOVER PERFORMANCE

9. Hover performance tests were conducted on the sixth year production aircraft at the conditions in table 1 using the tethered and free flight techniques described in appendix D. The 2-foot main wheel height in-ground effect (IGE) and the 100-foot main wheel height OGE tests were conducted at the 2302, 4120, and 9980 foot test sites in the normal utility (ESSS) configuration. Tip Mach number for these tests varied from 0.61 to 0.67. The data from these tests were compared with first year production aircraft hover data presented in USAAEFA Report No. 77-17 (ref 6, app A). The previous data was reanalyzed and a different curve was faired through both the 2-foot IGE and the 100-foot OGE data sets. When compared with the reanalyzed fairings, current test data reveal an increase in power required to hover IGE of approximately 5 percent and an increase of approximately 3 percent to hover OGE. Results are presented in figures 1 through 3, appendix E. These results compare favorably with previous OGE hover performance data with the ESSS fairings installed, USAAEFA Report No. 82-15-1 (ref 7, app A). No discernible compressibility trend was observed during this or any other previous testing. The hover performance results contained in this report should

be used to define the hover performance of a UH-60A in the normal utility (ESSS) configuration. The reanalyzed fairings for the first year production aircraft should be used to define the hover performance of a UH-60A in the normal utility configuration.

10. The standard day OGE hover ceiling at the primary mission gross weight of 16,455 pounds (app E) was 11,224 feet Hp using IRP available from USAAEFA Report No. 77-17 (ref 6, app A). At 4000 feet Hp on a 35°C day, the OGE hover maximum gross weight was 17,593 pounds with IRP. At the hover performance guarantee conditions of 95 percent IRP, 4700 feet Hp and 35°C, the OGE hover capability was 16,526 pounds.

LEVEL FLIGHT PERFORMANCE

11. Level flight performance tests were conducted at the conditions listed in table 1 to determine power required and fuel flow for airspeeds, altitudes, gross weights, and rotor speeds throughout a portion of the operational envelope of the sixth year production aircraft. Test data from USAAEFA Report No. 81-16 (ref 8, app A) was used to supplement the 258 rpm $N_R/\sqrt{\theta}$ data base. Techniques used in obtaining and analyzing level flight performance data are described in detail in appendix D. The data were obtained and analyzed in ball-centered flight and corrected for estimated drag of external test instrumentation and instrumentation electrical load.

12. Nondimensional test results are presented in figures 4 through 31, appendix E. The test data indicate power required generally increases with increasing $N_R/\sqrt{\theta}$. Trends at $N_R/\sqrt{\theta}$ above 258 rpm are not consistent with those of the first year production aircraft presented in USAAEFA Report No. 77-17 (ref 6, app A). The exponential increase in power required with increasing $N_R/\sqrt{\theta}$ as predicted by theory and as observed for the first year production aircraft (ref 6), was not evident for the sixth year production aircraft. Specific differences in power required between the first and sixth year aircraft throughout the $N_R/\sqrt{\theta}$ range tested did not produce a constant change in F_e (ΔF_e) between the two aircraft at all conditions. Comparing the normal utility and normal utility (ESSS) configurations at $N_R/\sqrt{\theta}$ of 258 rpm indicates a ΔF_e of approximately 5 ft^2. This difference was summarized as:

6th yr prod A/C = 1st year prod A/C + ESSS fairings (2.5 ft^2)
+ M-130 & AN/ALQ-144(V) brackets (1.5 ft^2)
+ external drag differences (1 ft^2)

Additional testing should be conducted in forward flight to investigate the inconsistencies in power required as a function of $N_R/\sqrt{\theta}$ between the first and sixth year production aircraft. The ΔF_e of the ESSS fairings was documented in USAAEFA Report No. 82-15-1 (ref 7), the M-130 and AN/ALQ-144(V) mounting brackets in paragraph 15, and the external configuration differences affecting drag between the two aircraft are depicted in the photographs in appendix B. Dimensional level flight test results are presented in figures 32 through 59, appendix E. Inherent sideslip, presented in figures 60 and 61, was developed from the resultant angle of sideslip associated with ball-centered flight during level flight performance testing (figs. 62 through 66). The data indicate that in ball-centered flight, sideslip increases to the right with increasing C_T. These results show the sixth year production UH-60A to fly with more inherent right sideslip when compared with previous test results (ref 6, app A) especially at higher C_T's.

13. Tests were conducted to ascertain the ΔF_e with sideslip for a range of C_T's and the data are presented in figure 67, appendix E, for both the normal utility and normal utility (ESSS) configurations. Results are independent of airspeed and $N_R/\sqrt{\theta}$, but vary with C_T. The data indicate that minimal F_e occurs between 4.5 and 7 degrees left sideslip depending upon C_T. Coordinated flight throughout the tested level flight airspeed envelope of the UH-60A results in a maximum left sideslip of approximately 1 degree.

14. Level flight performance testing on the UH-60A at different dimensional conditions that yield the same nondimensional condition have not produced consistent results. Stabilator position has been suspected to be a contributing factor to this discrepancy because the nondimensional parameters do not account for indicated airspeed which is the dimensional parameter that determines stabilator position in stabilized, ball-centered level flight at a given $C_T-\mu$ combination. Limited testing was conducted to determine the stabilator position effect on power required for level flight. Results are presented in figures 68 through 70, appendix E in the form of change in power coefficient as a function of deviation of stabilator from the programmed schedule position. Results vary with both C_T and μ The data show that stabilator trailing edge (TE) up movement produced an increase in power required and TE down movement produced a decrease. Sufficient data were gathered during the stabilator investigation to perform an analysis of the stabilator effect at a C_T of 0.009 at various values of μ. No stabilator corrections have been made to the level flight data presented in appendix E, however, a limited analysis was performed using the data available at a C_T of approximately 0.009, for

example figures 40 and 42. The fairing on these two figures represents a normalization process based to a large extent on the two data sets because of their proximity, but is also influenced by cross fairing C_T, C_P, μ, and $N_R/\sqrt{\theta}$ of all the tests. The fairing can be made to better approximate the data by determining for a specific μ, the difference in indicated airspeed due to the different dimensional conditions representing the data and normalized fairing. This difference can then be converted into a change in stabilator position and consequently a change in power required. For example, the fairing in figure 42 represents an altitude greater than that represented by the data and denotes less power required than the data. Decreasing altitude increases indicated airspeed for the same μ, which positions the stabilator more TE up, thereby requiring a corresponding increase in power to maintain level flight raising the faired line. Applying the stabilator correction, however, accounts for less than half of the difference in power required between the two C_T data sets, after equating both to a nominal C_T. Therefore, regardless of the limited amount of data accumulated on stabilator effects and their consequences on power required, it is assumed that other unexplained aerodynamic effects preclude accurate nondimensionalizing of level flight performance. If these differences are to be fully explained, further stabilator tests at a range of airspeeds throughout the C_T envelope, and a study undertaken to identify remaining differences complemented with verification testing should be accomplished.

15. Testing was accomplished earlier on a first year production aircraft for inclusion in this report of the performance change associated with the installation of the AN/ALQ-144(V) infrared countermeasures set and M-130 chaff dispenser. Level flight performance test results are presented in figures 54 through 59, appendix E. The mounting brackets for the AN/ALQ-144(V) and M-130 sets produce 1.5 ft^2 of F_e. Installation of the AN/ALQ-144(V) and M-130 sets increases F_e by an additional 0.5 ft^2. The slightly high fairings at C_T of 0.009 could be lowered if the stabilator correction described in paragraph 14 was applied.

AIRSPEED CALIBRATION

16. The standard ship's airspeed system on the sixth year production aircraft was calibrated in level flight. A calibrated T-28 pace aircraft and a calibrated trailing bomb were used to determine the position error. The position error of the ship's airspeed system ia presented in figure 71, appendix E. In level flight, airspeed position error varied from -8 knots at 35 knots indicated airspeed (KIAS) to +3 knots at 160 KIAS. This represents a

decrease in position error of almost 2 knots from the position error determined with the prototype production airspeed system (ref 9, app A). Additional testing to determine the airspeed position error of the production airspeed system over a broader range of flight conditions should be conducted.

CONCLUSIONS

17. Based on this evaluation, the following conclusions can be drawn about the performance of the sixth year production UH-60A in the normal utility (ESSS) configuration:

a. Power required to hover was increased compared to the first year production normal utility configured UH-60A (para 9).

b. Increased requirement for power was measured generally as a result of increasing referred rotor speed ($N_R/\sqrt{\theta}$), although the exponential increase predicted by theory was not realized (para 12).

c. Drag in level flight increased by 5 square feet (ft^2) of equivalent flat plate area (F_e) compared to the normal utility configured UH-60A at a $N_R/\sqrt{\theta}$ of 258 rpm (para 12).

d. Drag of the mounting brackets for the AN/ALQ-144(V) and M-130 sets in level flight was 1.5 ft^2 of F_e, and installation of the sets increased total F_e by an additional 0.5 ft^2 (para 15).

e. The difference in stabilator position accounts for less than half the difference in level flight power required when different dimensional conditions produce the same nondimensional condition for a C_T of 0.009 (para 14).

RECOMMENDATIONS

18. The following recommendations are made:

a. Hover performance determined during this evaluation should be used for a UH-60A in the normal utility (ESSS) configuration (para 9).

b. Reanalyzed hover performance fairings of USAAEFA Project No. 77-17 produced during this evaluation should be used for a UH-60A in the normal utility configuration (para 9).

c. Additional testing should be conducted in forward flight to investigate the cause of the variations in power required as a function of referred rotor speed between the first and sixth year production aircraft (para 12).

d. Further stabilator testing is necessary at a range of airspeeds throughout the thrust coefficient envelope of the UH-60A if it's effect on level flight power required is to be fully documented (para 14).

e. A study, complemented by testing for verification, should be undertaken to identify unexplained aerodynamic effects that preclude accurate nondimensionalizing of level flight performance of the UH-60A (para 14).

f. Additional testing throughout a range of flight conditions should be conducted to further evaluate the differences in airspeed position error between the prototype system presented in USAAEFA Project No. 82-09 and the current airspeed system incorporated in the sixth year production aircraft (para 16).

APPENDIX A. REFERENCES

1. Letter, AVRADCOM, DRDAV-DI, 30 September 1983, subject: Airworthiness and Flight Characteristics (A&FC) Test of a Sixth Year Production UH-60A, USAAEFA Project No. 83-24.

2. Prime Item Development Specification, Sikorsky Aircraft Division, *DARCOM-CP-2222-S1000F Part I*, 15 December 1981.

3. Technical Manual, TM55-1520-237-10, *Operator's Manual, UH-60A Helicopter, Headquarters Department of the Army*, 21 May 1979 with change 22 dated 2 September 1983.

4. Letter, AVSCOM, DRSAV-E, 21 February 1984, subject: Airworthiness Release for the UH-60A Black Hawk Helicopter, S/N 82-23748, to Conduct an Airworthiness and Flight Characteristics (A&FC) Test of a Sixth Year Production UH-60A, USAAEFA Project No. 83-24.

5. Test Plan, USAAEFA Project No. 83-24, *Airworthiness and Flight Characteristics Test of a Sixth Year Production UH-60A*, December 1983.

6. Final Report, USAAEFA Project No. 77-17, *Airworthiness and Flight Characteristics Evaluation UH-60A (Black Hawk) Helicopter*, September 1981.

7. Final Report, USAAEFA Project No. 82-15-1, *UH-60A External Stores Support System Fixed Provision Fairings Drag Determinations*, May 1984.

8. Final Report, USAAEFA Project No. 81-16, *UH-60A Expanded Gross Weight and Center of Gravity Evaluation*, Unpublished.

9. Final Report, USAAEFA Project No. 82-09, *Preliminary Airworthiness Evaluation of UH-60A with an Improved Airspeed System*, April 1983.

10. Technical Manual, TM55-1520-237-23-2, *Aircraft General Information Manual, UH-60A Helicopter*, Headquarters Department of the Army, 29 December 1978.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The Sikorsky UH-60A (Black Hawk) is a twin-turbine engine, single main rotor helicopter capable of transporting 11 combat troops plus a crew of three. It is equipped with three nonretractable conventional wheel-type landing gear. A movable horizontal stabilator is located on the lower portion of the tail rotor pylon. The main and tail rotors are both four-bladed with a capability of manual main rotor blade and tail pylon folding. The cross-beam tail rotor with composite blades is attached to the right side of the pylon and is canted 20 degrees upward from the horizontal. A complete description of the aircraft is contained in the operator's manual (ref 3, app A) and the aircraft general information manual (ref 10).

2. Two helicopters were used in this evaluation, first year (USA S/N 77-22716) and sixth year (USA S/N 82-23748) production aircraft. The following photographs 1 through 12 illustrate the configuration differences between the two aircraft in their respective normal utility configurations.

EXTERNAL STORES SUPPORT SYSTEM (ESSS) FIXED PROVISION FAIRINGS

3. The sixth year production aircraft is equipped with provisions for incorporating the ESSS. With the system removed, aerodynamic fairings are installed (photo 1). The weight of the integral airframe fixed provisions is 123 pounds, the removable provisions are 8 pounds, and the total is included in the aircraft basic weight. The first year production aircraft does not include provisions for the ESSS (photo 2).

COUNTERMEASURE PROVISIONS

4. The sixth year production aircraft is equipped with the AN/ALQ-144(V) Infrared (IR) countermeasures set and an M-130 chaff/flare dispenser. These units were removed for testing, but the brackets supporting them remained (photos 3 and 4). The first year production aircraft does not incorporate these countermeasure devices. However, aircraft USA S/N 77-22716 (first year production aircraft) was tested with the brackets added and with the countermeasure devices installed to determine their effect on level flight performance.

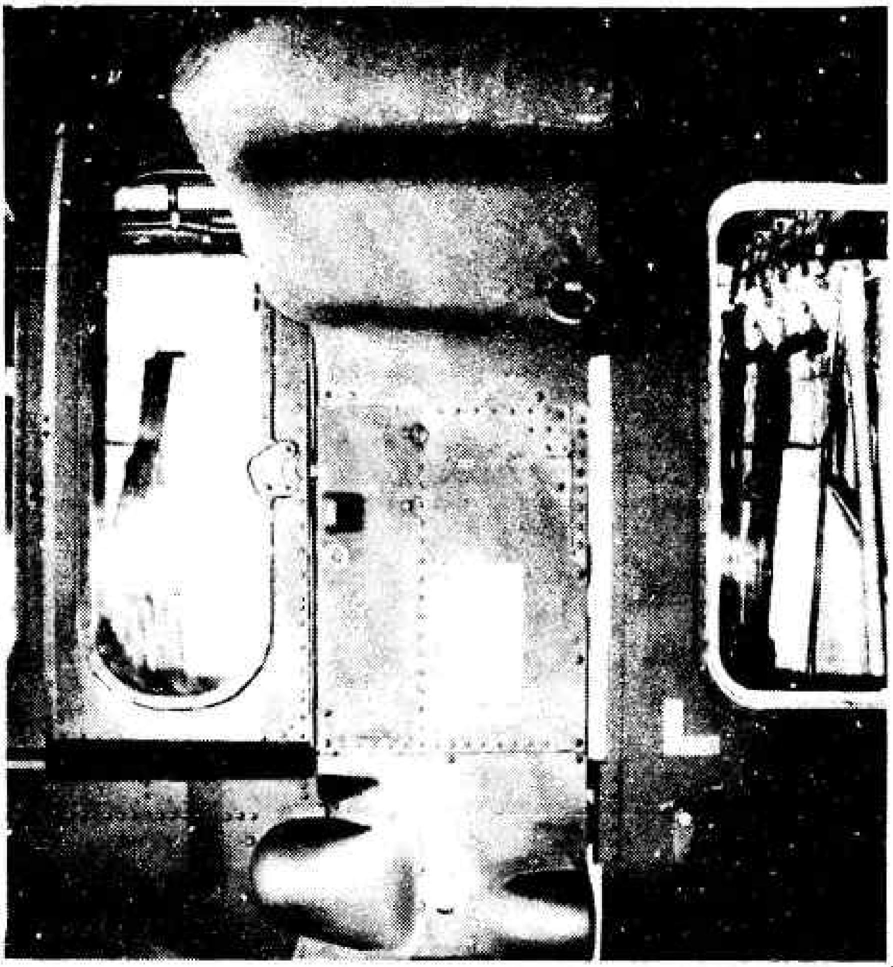


Photo 1. ESSS Fairing Installation, Left Side (Normal Utility (ESSS) Configuration, Sixth Year Production Aircraft)

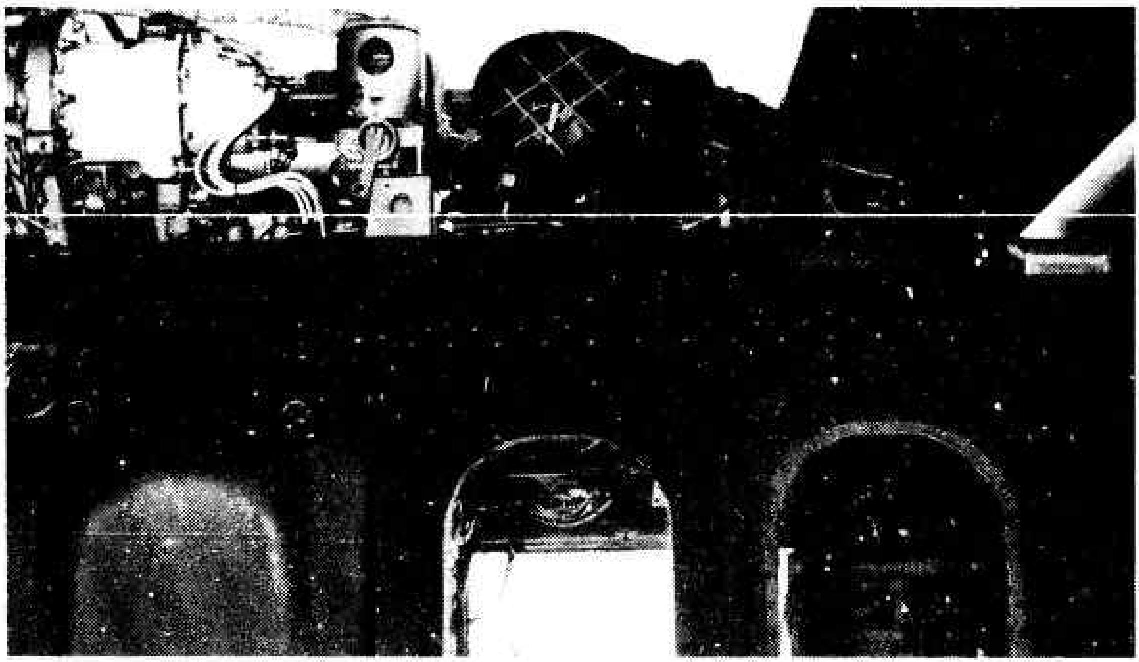


Photo 2. Right Side (Normal Utility Configuration, First Year Production Aircraft)

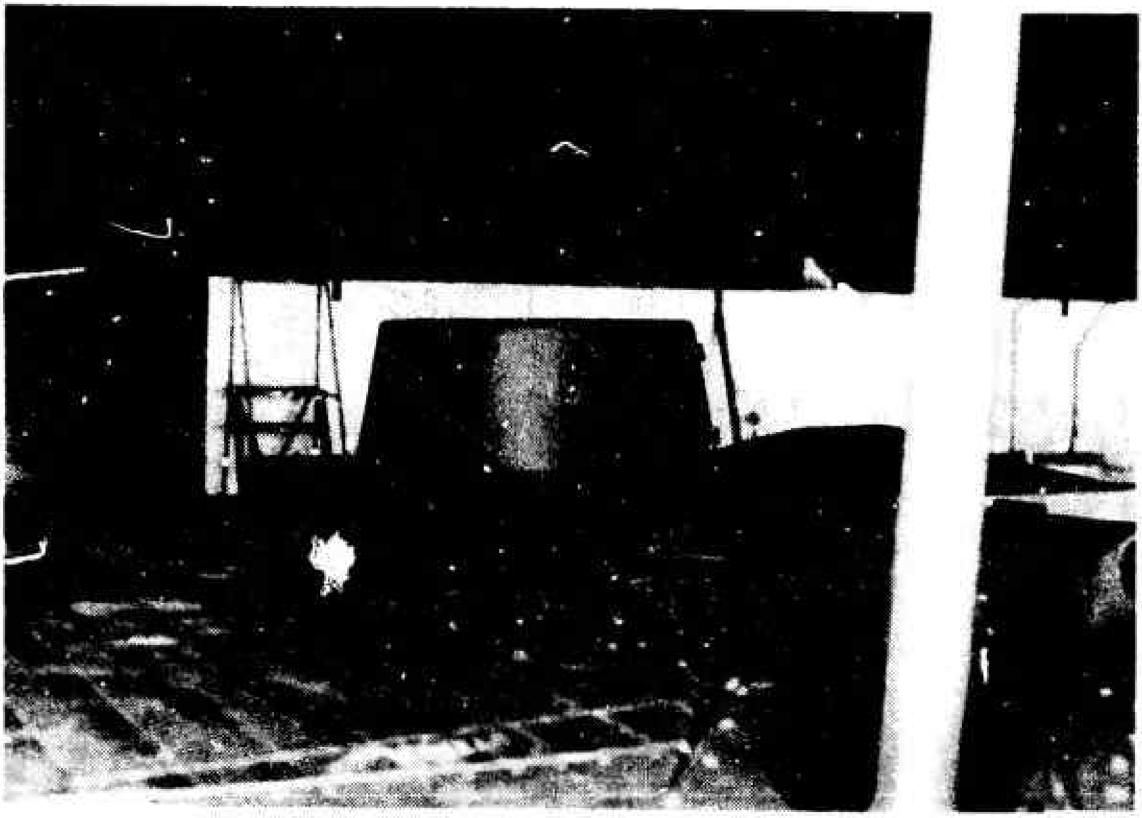


Photo 3. AN/ALQ-144(V) IR Countermeasure Bracket
(Sixth Year Production Aircraft)

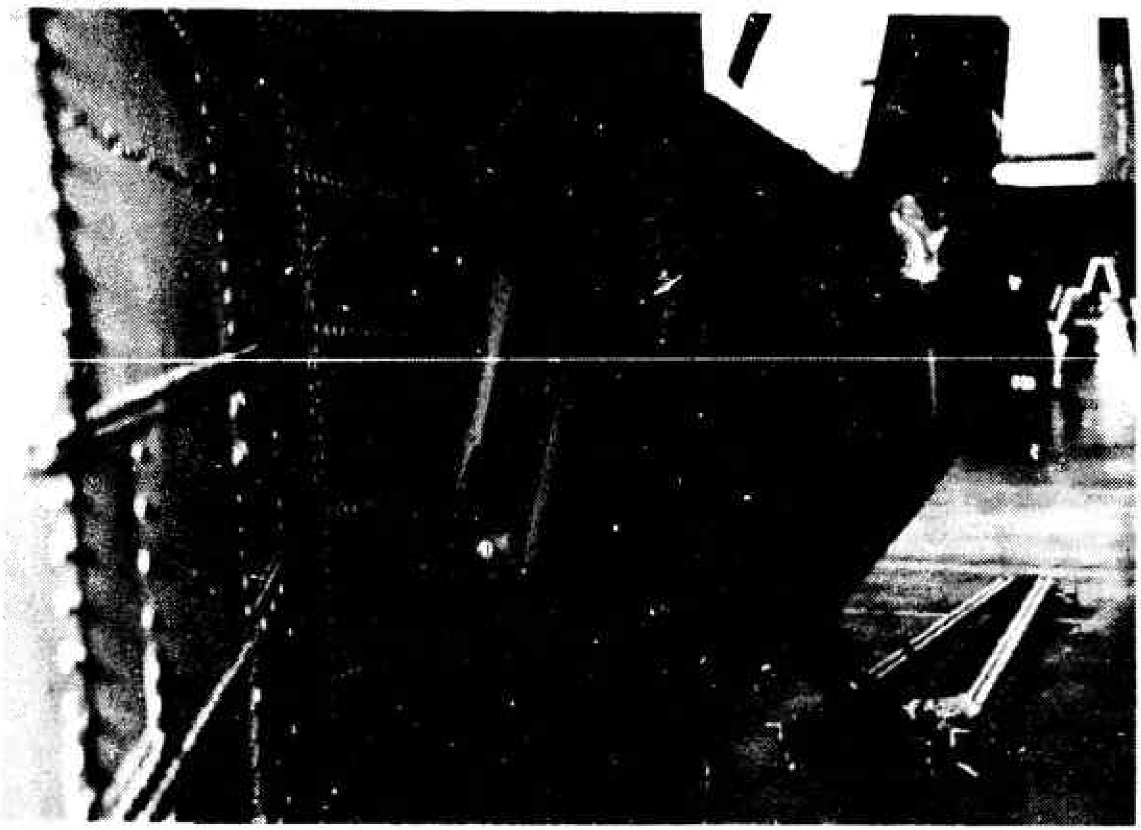


Photo 4. M-130 General Purpose Dispenser Bracket
(Sixth Year Production Aircraft)

AIRSPEED/STABILATOR MODIFICATIONS

5. The airspeed/stabilator system on both test aircraft incorporated the modifications developed during USAAEFA Report No. 82-09. The first year production aircraft incorporated the development, or prototype production system, while the sixth year aircraft included the contemporary production system. Three changes were incorporated in the pitot-static pressure systems and two changes in the electrical circuit to the stabilator amplifiers of the stabilator system. Major changes from the original production version incorporated in both aircraft were: reorienting the pitot-static tube 20 degrees outboard and 3 degrees down, venting the vertical speed indictor static source from the pitot-static tube to the cabin, damping the airspeed indicator 0.4 seconds, increasing the damping of the stabilator to 3.0 seconds, and reducing the collective bias of the stabilator schedule at high collective settings. The mount to reorient the pitot-static tube of the first year test aircraft varied from the production mount on the sixth year aircraft in height (photos 5 and 6).

MISCELLANEOUS

6. The bifilar absorbers of the sixth year production aircraft are redesigned in comparison to the first year production absorbers (photos 7 and 8).

7. Sixth year production aircraft are equipped with the rotor deicing system, while the first year production aircraft was not. The deice system incorporates main and tail rotor deicing capabilities. Photo 9 shows the main rotor slip ring and distributor assembly of the deice system. The main rotor hubs of both test aircraft were adapted with a slip ring assembly for instrumentation purposes (photo 10). Other drag producing components of the deice system are the ice detector probe located on the right engine nacelle (photo 11) and the outside air temperature sensor located on the nose of the aircraft in front of the center windshield (photo 12).

ENGINES

8. The primary power plants for the UH-60A helicopter are General Electric T700-GE-700 front drive turboshaft engines, rated at 1553 shaft horsepower (SHP) at a power turbine speed of 20,900 revolutions per minute (rpm) (sea level, standard day installed). The engines are mounted in nacelles on either side of the main transmission. Each engine has four modules: cold section, hot

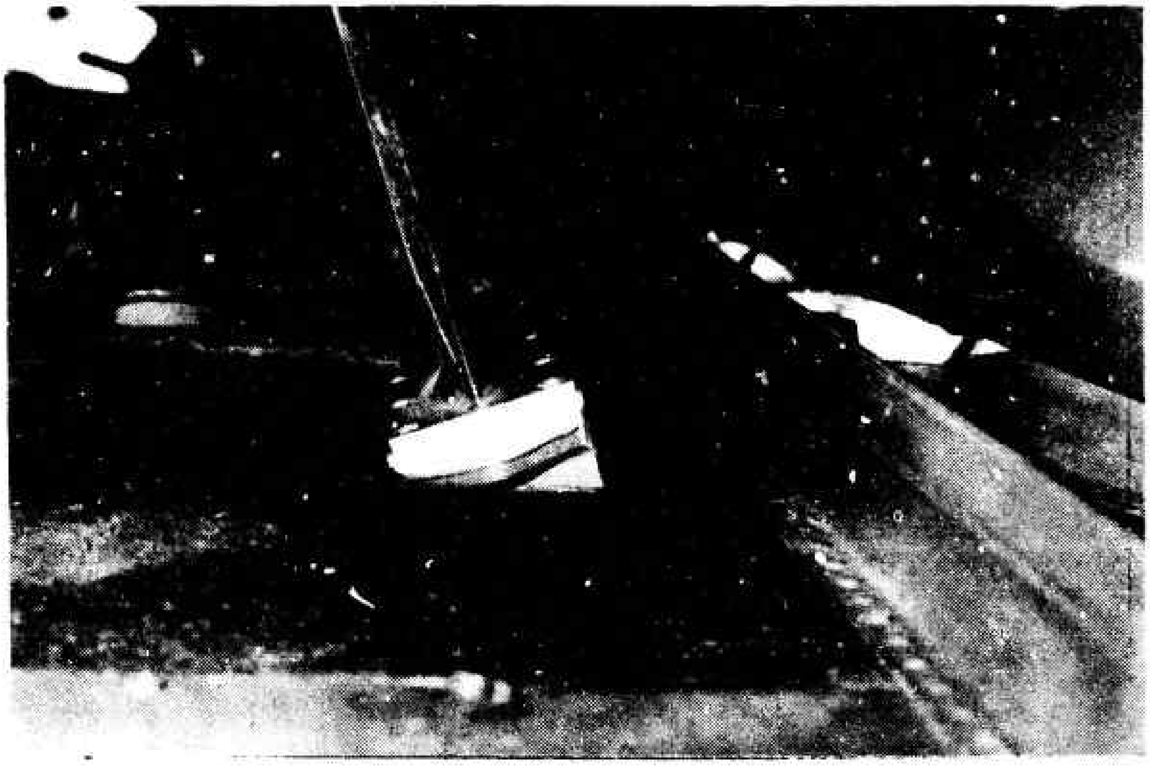


Photo 5. Modified Production Pitot-Static System Mount (Normal Utility (ESSS) Configuration)

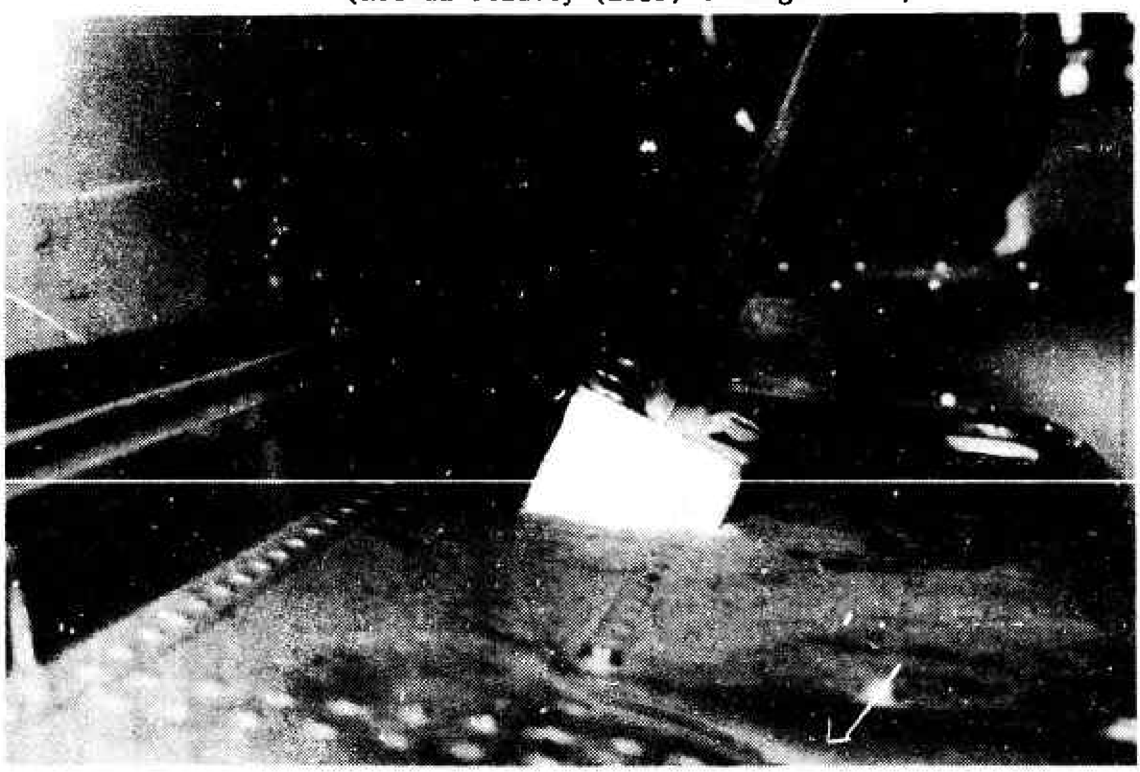


Photo 6. Development Mount for Modified Production Pitot-Static System Installed on Aircraft USA S/N 77-22716 During this Testing

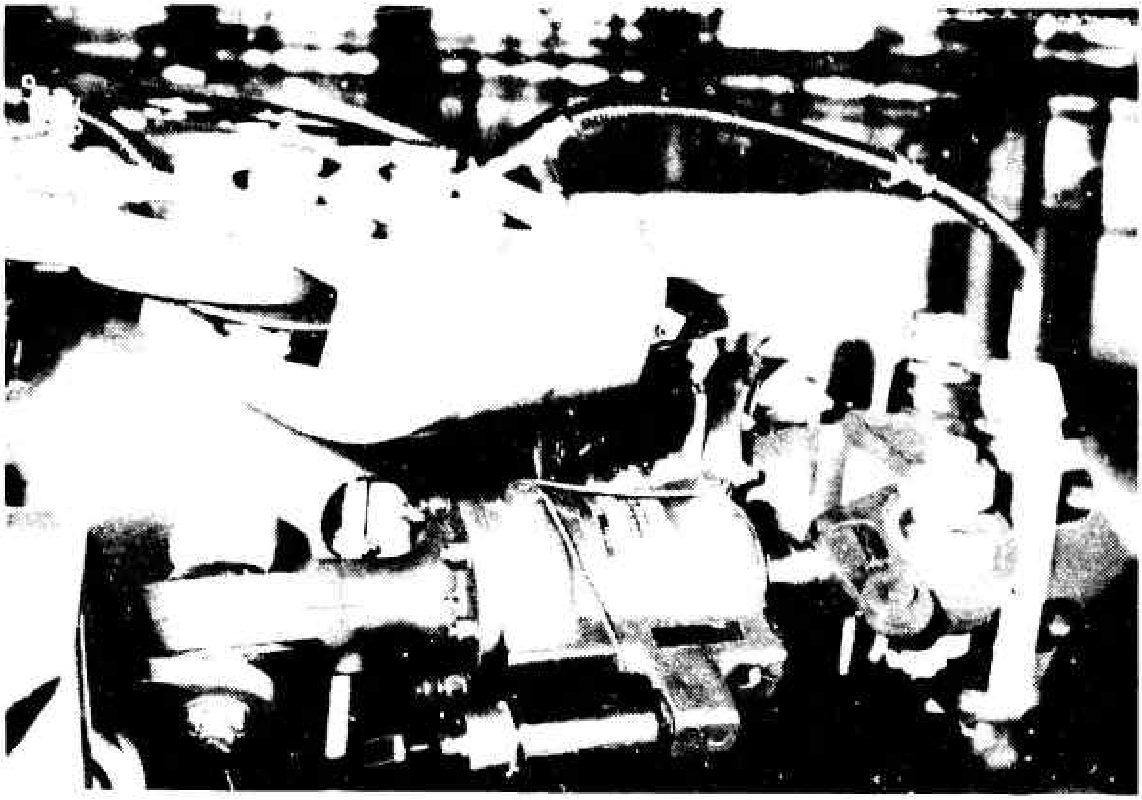


Photo 7. Bifilar Absorber (Sixth Year Production Aircraft)

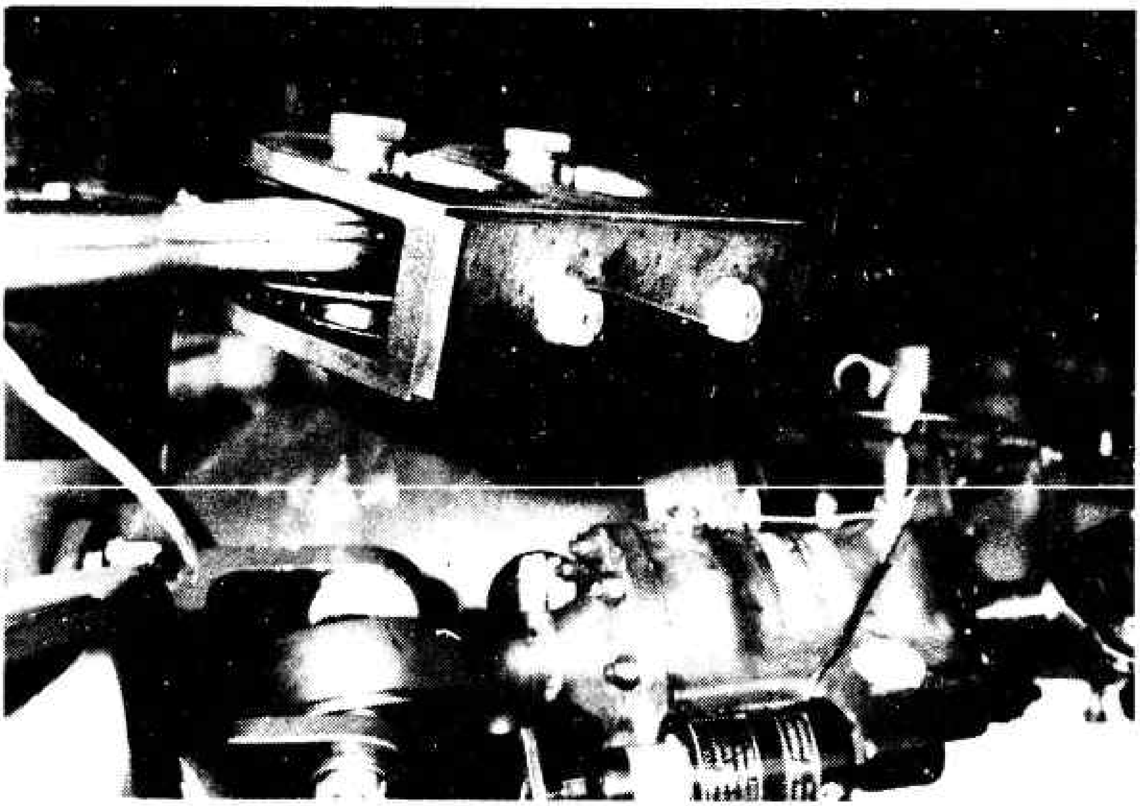


Photo 8. Bifilar Absorber (First Year Production Aircraft)

Photo 9. Main Rotor Hub (Sixth Year Production Aircraft)

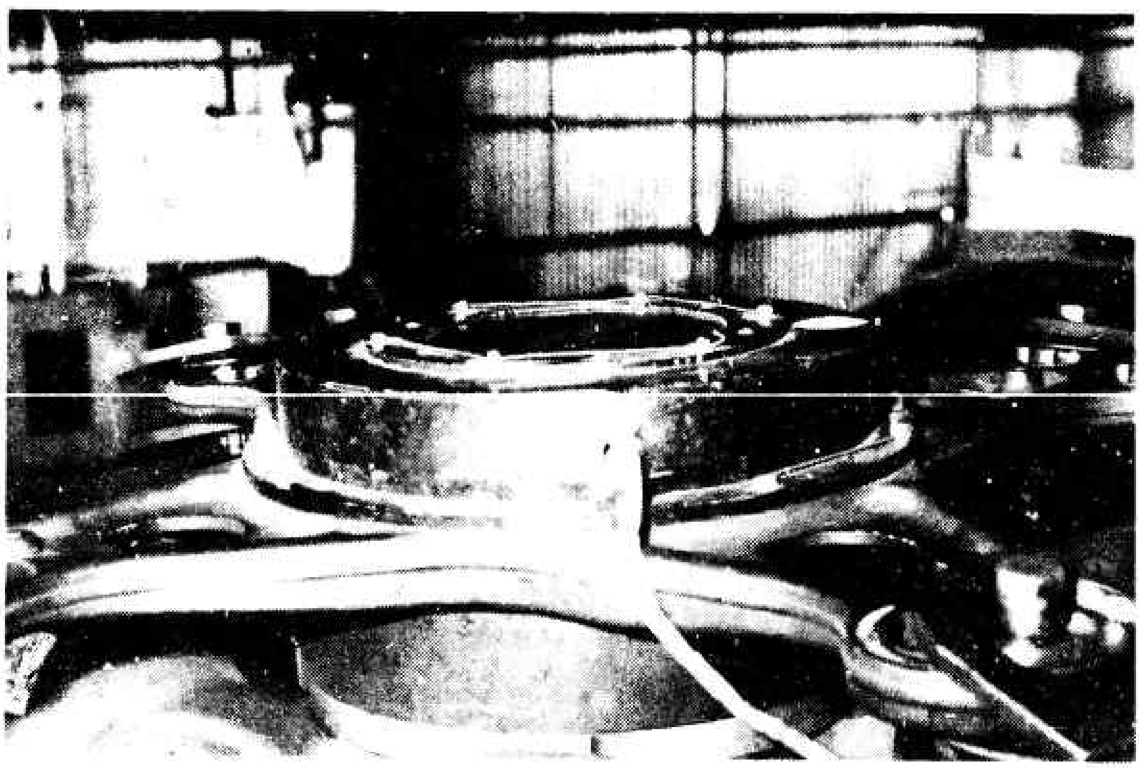


Photo 10. Instrumented Main Rotor Hub (Both Test Configurations)

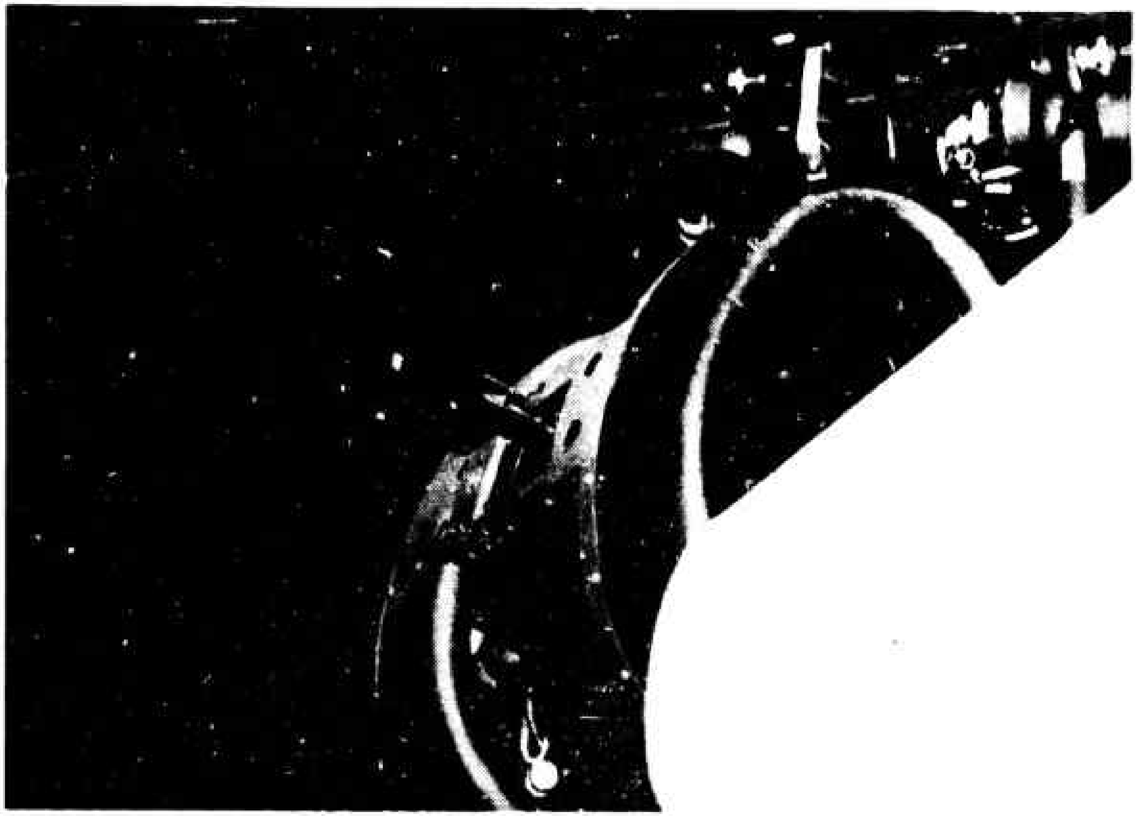


Photo 11. Ice Detector Probe, Right Side (Sixth Year Prcduction Aircraft)

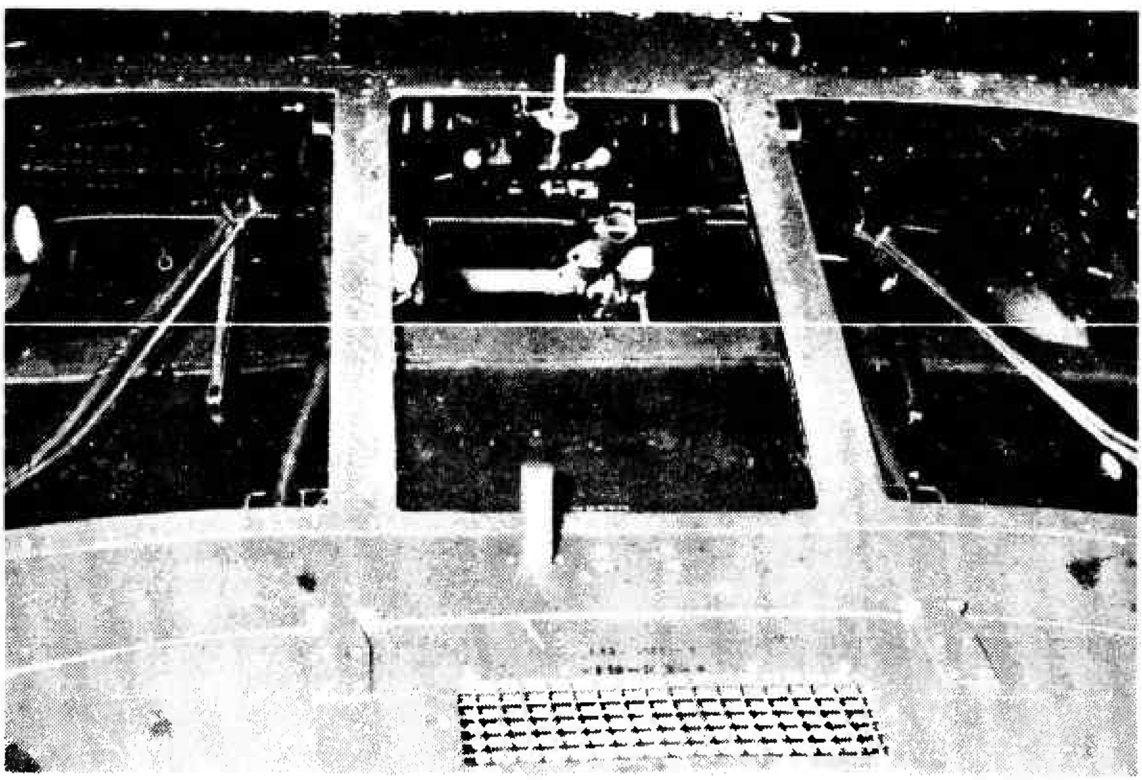


Photo 12. Deice System Temperature Sensor (Sixth Year Production Aircraft)

section, power turbine section, and accessory section. Design features include an axialcentrifugal flow compressor, a through-flow combustor, a two-stage air-cooled high pressure gas generator turbine, a two-stage uncooled power turbine, and self contained lubrication and electrical systems. Pertinent engine data are shown below.

Model	T700-GE-700
Type	Turboshaft
Rated power	1553 SHP installed at sea level, standard-day static conditions at 20,900 rpm
Compressor	Five axial stages, 1 centrifugal stage
Combustion chamber	Single annular chamber with axial flow
Gas generator stages	2
Power turbine stages	2
Direction of engine rotation (aft looking fwd)	Clockwise
Weight (dry)	415 pounds max
Length	47 in.
Maximum diameter	25 in.
Fuel	MIL-T-5624 grade JP-4 or JP-5

BASIC AIRCRAFT INFORMATION

9. General data of the sixth year production UH-60A helicopter are as follows:

Gross Weight

Empty weight	Approximately 10,750 pounds
Primary Mission gross weight	16,455 pounds
Fuel capacity (measured)	359 gallons

Main Rotor

Number of blades	4
Diameter	53 ft, 8 in.
Blade chord	1.73/1.75 ft
Blade twist	-18 deg (equivalent)

Blade tip sweep	20 deg aft
Blade area (one blade)	46.7 sq ft
Airfoil	
section (root to tip designation)	SC1095/SC1095R8
thickness (percent chord)	9.5 percent
Main rotor mast tilt (forward)	3 deg

Tail Rotor

Number of blades	4
Diameter	11 ft
Blade chord	0.81 ft
Blade twist (equivalent linear)	-18 deg
Blade area (one blade)	4.46 sq ft
Airfoil	
section (root to tip designation)	SC1095/SC1095R8
thickness (percent chord)	9.5 percent
Shaft cant angle (upward)	20 deg

Gear Ratios

Main Transmission	Input RPM	Output RPM	Ratio	(Teeth)
Input bevel	20,900.0	5747.5	3.6364	(80/22)
Main bevel	5747.5	1206.3	4.7647	(81/17)
Planetary	1206.3	257.9	4.6774	$\frac{(228 + 62)}{62}$
Tail takeoff	1206.3	4115.5	0.2931	(34/116)
Accessory bevel (generator)	5747 5	11,805.7	0.4868	(37/76)
Accessory spur (hydraulics)	11,805.7	7186.1	1.6429	(92/56)
Intermediate Gearbox	4115.5	3318.9	1.2400	(31/25)
Tail Gearbox	3318.9	1189.8	2.7895	(53/19)

Overall

Engine to main rotor	20,900.0	257.9	81.0419
Engine to tail rotor	20,900.0	1189.8	17.5658
Tail rotor to main rotor	1189.8	257.9	4.6136

APPENDIX C. INSTRUMENTATION

GENERAL

1. The test instrumentation was installed, calibrated and maintained by the US Army Aviation Engineering Flight Activity. A test boom, with a swiveling pitot-static tube and angle of attack and sideslip vanes, was installed at the nose of the aircraft. Equipment required for specific tests was installed when needed. Data was obtained from calibrated instrumentation and displayed or recorded as indicated below.

Pilot Panel

Airspeed (boom)
Airspeed (ship)*
Altitude (boom)
Altitude (ship)*
Altitude (radar)*
Rate of climb*
Rotor speed (sensitive-digital)
Engine torque* **
Turbine gas temperature* **
Power turbine speed (N_p)* **
Gas producer speed (N_g)* **
Control position
    Longitudinal
    Lateral
    Directional
    Collective
Horizontal stabilator position*
Center of gravity (cg) lateral acceleration (sensitive)
Angle of sideslip
Tether cable angles
    Longitudinal
    Lateral

Copilot Panel

Event switch
Airspeed*
Altitude*
Rotor speed*
Engine torque* **
Ballast cart control
Ballast cart position
Cable tension
Fuel remaining* **

*Ship's system/not calibrated
**Both engines

Engineer Panel

Preaaure altitude
Ambient pressure
Engine Fuel flow**
Engine Fuel used**
APU fuel used
Total air temperature
Instrumentation controls
Time code display
Run number
Event awitch

2. Data parameters recorded on board the aircraft and via telemetry include the following:

Digital (PCM) Data Parameters

Airspeed (boom)
Altitude (boom)
Airspeed (ship's)
Altitude (ship's)
Total air temperature
Rotor speed
Gas generator speed**
Power turbine speed**
Engine fuel flow**
Engine fuel used**
Engine fuel temperature**
Engine output shaft torque**
Turbine gas temperature**
APU fuel used
Main rotor shaft torque
CG lateral acceleration (aenaitive)
Tether cable tension
Tether cable angle
    Longitudinal
    Lateral
Stabilator position
Movable ballaat location
Control position
    Longitudinal
    Lateral
    Directional
    Collective
Attitude
    Pitch
    Roll
    Yaw

**Both engines

Angular Rate
  Pitch
  Roll
  Yaw
Tail rotor shaft torque
Tail rotor impressed pitch (Blade angle at 0.75 blade span)
Angle of sideslip
Angle of attack
Time of day
Run number
Pilot event
Engineer event

AIRSPEED CALIBRATION

3. The standard ship's airspeed system and test boom airspeed system were calibrated in level flight. The ground speed course, a calibrated T-28 pace aircraft and a calibrated trailing bomb (finned pitot-static system) were used to determine the position error. The position error of the boom airspeed system is presented in figure 1.

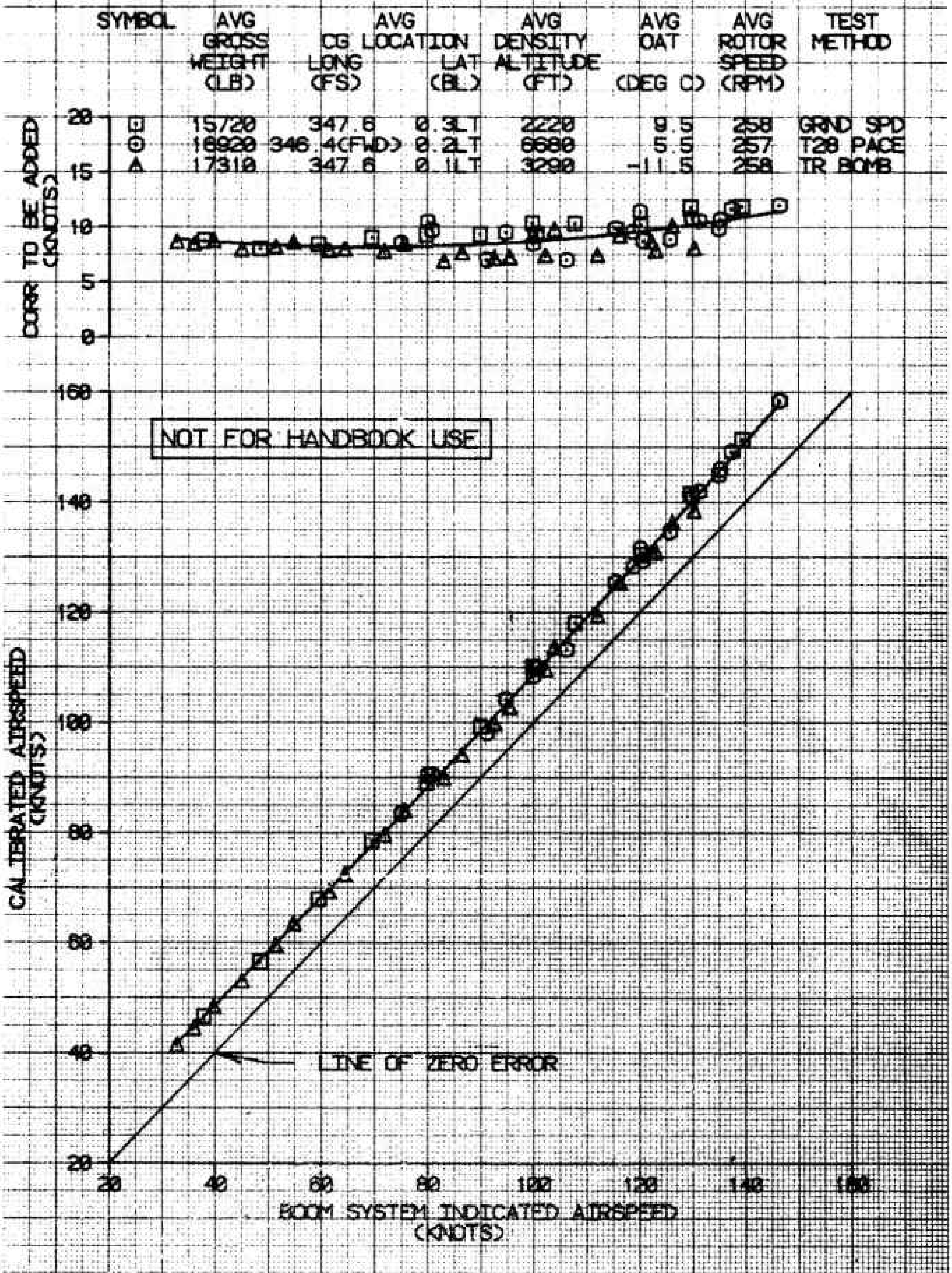
WEATHER STATION

4. A portable weather station was used during tethered hover tests. The weather station equipment included an anemometer to measure wind speed and direction at selected heights up to 100 feet above ground level. A sensitive temperature gage and barometer were utilized to measure ambient temperature and atmospheric pressure, respectively.

LOAD CELL

5. A calibrated load cell was incorporated with the ship's cargo hook to measure cable tension and accelerometers were used to measure longitudinal and lateral cable angles for tethered hover tests. Indicators were installed in the cockpit to display cable tension and cable angle measured with respect to the ground.

FIGURE 1
BOOM SYSTEM AIRSPEED CALIBRATION IN LEVEL FLIGHT
UH-60A USA S/N 82-23748



APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

AIRCRAFT RIGGING

1. A flight controls engineering rigging check was performed on the main and tail rotors to insure compliance with established limits. The stabilator control system was adjuated to conform as close aa possible to the modified production schedule to prevent improper drag characteristics effecting level flight performance.

AIRCRAFT WEIGHT AND BALANCE

2. The aircraft was weighed in the instrumented configuration with all fuel drained and full oil prior to the start of the Airworthiness and Flight Characteristics program. The initial weight of the sixth year production aircraft was 12,000 pounds with the longitudinal center of gravity (cg) located at fuselage station (FS) 352.2 with the cg of the empty ballast cart located at FS 301. The fuel cells and an external sight gage were alao calibrated. The measured fuel capacity using the gravity fueling method was 359 gallons. The fuel weight for each test flight was determined prior to engine start and after engine shutdown by using the external sight gage to determine the volume and measuring the specific gravity of the fuel. The calibrated cockpit fuel totalizer indicator was used during the test and at the end of each test compared with the sight gage readinga. Aircraft cg was controlled by a movable ballast system which was manually positioned to maintain a constant cg whila fuel was burned. The movable ballast system was a cart (2000-pound capacity) attached to the cabin floor by rails and driven by an electric screw jack with a total longitudinal travel of 72.3 inches.

PERFORMANCE

General

3. Helicopter performance was generslized through the use of nondimensional coefficients as follows using the 1968 US Standard Atmosphere:

a. Coefficient of Power (C_P):

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

b. Coefficient of Thrust (C_T):

$$C_T = \frac{GW + \text{CABLE TENSION}}{\rho A(\Omega R)^2} \quad (2)$$

c. Advance Ratio (μ):

$$\mu = \frac{V_T\ (1.6878)}{\Omega R} \quad (3)$$

Where:

SHP = Engine output shaft horsepower (total for both engines)

ρ = Ambient air density (lb-sec^2/ft^4) = $\rho_o \left[\frac{\delta}{\theta}\right]$

ρ_o = 0.0023769 (lb-sec^2/ft^4)

δ = Pressure ratio = $\frac{P_a}{P_{ao}}$

P_a = Ambient air pressure (in.-Hg)

P_{ao} = 29.92126 in.-Hg

θ = Temperature ratio = $\frac{OAT + 273.15}{288.15}$

OAT = Ambient air temperature (°C)

A = Main rotor disc area = 2262 ft^2

Ω = Main rotor angular velocity (radians/sec)

R = Main rotor radius = 26.833 ft

GW = Gross weight (lb)

$$V_T = \text{True airspeed (kt)} = \frac{V_E}{1.6878\sqrt{\rho/\rho_0}}$$

1.6878 = Conversion factor (ft/sec-kt)

V_E = Equivalent airspeed (ft/sec) =

$$\left\{ \frac{7(70.7262\ P_a)}{\rho_0} \left(\left[\left(\frac{Q_c}{P_a} \right) +1 \right]^{2/7} -1 \right) \right\}^{1/2}$$

70.7262 = Conversion factor (lb/ft^2-in.-Hg)

Q_c = Dynamic pressure (in.-Hg)

At the normal operating rotor speed of 257.9 revolutions per minute (rpm) (100%), the following constants may be used to calculate C_P and C_T:

$\Omega R = 724.685$

$(\Omega R)^2 = 525{,}168.15$

$(\Omega R)^3 = 380{,}581{,}411.2$

4. The engine output shaft torque was determined by use of the engine torque sensor. The power turbine shaft contains a torque sensor tube that measures the total twist of the shaft. A concentric reference shaft is secured by a pin at the front end of the power turbine drive shaft and is free to rotate relative to the power turbine drive shaft at the rear end. The relative rotation is due to transmitted torque, and the resulting phase angle between the reference teeth on the two shafts is picked up by the torque sensor. This torque sensor for both engines was calibrated in a test cell by the engine manufacturer. The output from the engine torque sensor was recorded on the onboard data recording system. The output SHP was determined from the engine's output shaft torque and rotational speed by the following equation.

$$SHP = \frac{Q(N_P)}{5252.113} \qquad (4)$$

Where:

Q = Engine output shaft torque (ft-lb)

N_p = Engine output shaft rotational speed (rpm)

5252.113 = Conversion factor (ft-lb-rev/min-SHP)

The output SHP required was assumed to include 13 horsepower for daylight operations of the aircraft electrical system, but was corrected for the effects of test instrumentation installation. A power loss of 1.82 horsepower was determined for electrical operation of the instrumentation. Reductions in power required were made for the effect of external instrumentation drag. This was determined by the following equation.

$$SHP_{instr\ drag} = \frac{F_e\ (\rho/\rho_o)(V_T)^3}{96254} \tag{5}$$

Where:

F_e = 0.833 ft^2 (estimated)

96254 = Conversion factor (ft^2-kt^3/SHP)

The nominal fuel temperature of 50°C for the cold weather test site and 55°C for remaining test sites was used in the determination of engine fuel conaumption.

Shaft Horsepower Available

5. The SHP available for the T700-GE-700 engine installed in the UH-60A was obtained from data received from US Army Aviation Systems Command and preaented in USAAEFA Report No. 77-17 (ref 6, app A). This data was calculated using the General Electric engine deck number 80024, dated 26 February 1981 with a power turbine shaft speed of 20,900 rpm. The installation losses used were based on 0.25 degree C engine inlet temperature rise in a hover, exhaust losses as obtained from the Sikorsky Aircraft Document Number SER-70410, Revision 2, dated 8 March 1979, inlet ram pressure recovery as obtained from the Sikorsky Prime Item Development Specification, and an inlet temperature rise in forward flight assuming an adiabatic rise referenced to ambient.

Hover Performance

6. Hover performance was obtained by the tethered hover technique. Additional free flight hover data were accumulated to verify the tethered hover data. All hover tests were conducted in winds of less than 3 knots. Tethered hover consists of restraining the helicopter to the ground by a cable in series with a load cell. An increase in cable tension, measured by the load cell, is equivalent to an increase in gross weight. Free flight hover tests consisted of stabilizing the helicopter at a desired height using the radar altimeter as a height reference. All hovering data were reduced to nondimensional parameters of C_P and C_T using equations 1 and 2, respectively, and grouped according to wheel height. A two segment fairing was used to more accurately represent the out-of-ground effect hover performance. Fairings of the same form used in this analysis were used in a reanalysis of the data representing the normal utility configured UH-60A (ref 6, app A) to yield a more indicative comparison. Summary hovering performance was then calculated from these nondimensional plots using the power available from reference 6.

Level Flight Performance

General:

7. Each speed power was flown in ball-centered flight by reference to a sensitive lateral accelerometer at a predetermined C_T and referred rotor speed $(N_R/\sqrt{\theta})$. To maintain the ratio of gross weight to pressure ratio constant, altitude was increased as fuel was consumed. To maintain $N_R/\sqrt{\theta}$ constant, rotor speed was decreased as temperature decreased. Power corrections for rate-of-climb and acceleration were determined (when applicable) by the following equations.

$$SHP_{R/C} = -\frac{(R/C_{TL})(GW)}{33{,}000(K_P)} \qquad (6)$$

$$SHP_{ACCEL} = -1.6098 \times 10^{-4}\left(\frac{\Delta V}{\Delta t}\right)(V_T)\ (GW) \qquad (7)$$

Where:

$$R/C_{TL} = \text{Tapeline rate of climb (ft/min)} = \left(\frac{\Delta H_P}{\Delta t}\right)\left(\frac{OAT + 273.15}{OAT_s + 273.15}\right)$$

$\dfrac{\Delta Hp}{\Delta t}$ = Change in pressure altitude per unit time (ft/min)

OAT_s = Standard ambient temperature at pressure altitude where $\dfrac{\Delta Hp}{\Delta t}$ was measured (°C)

Kp = 0.76

1.6098×10^{-4} = Conversion factor (SHP-sec/kt^2-lb)

$\dfrac{\Delta V}{\Delta t}$ = Change in airspeed per unit time (kt/sec)

A power correction to insure ball-centered test data complied with the inherent sideslip family of curves depicting the UH-60A in figures 60 and 61, appendix E, was determined from ΔF_e as a function of sideslip angle (fig. 67) and equation 5 rewritten as follows.

$$SHP_{s/s} = \frac{(\Delta F_{e\ in\ s/s} - \Delta F_{e\ B-C})\ (\rho/\rho_o)\ (V_T^3)}{96254} \tag{8}$$

Where:

$\Delta F_e{}^*{}_{in\ s/s}$ = Change in equivalent flat plate area based on UH-60A inherent sideslip.

$\Delta F_e{}^*{}_{B-C}$ = Change in equivalent flat plate area based on the sideslip angle measured in ball-centered flight.

*Based on change in engine shaft horsepower.

Power required for level flight at the test day conditions was determined using the following equation.

$$SHP_t = SHP + SHP_{R/C} + SHP_{ACCEL} + SHP_{s/s} - SHP_{instr\ drag} - 1.82 \tag{9}$$

8. Test day level flight data was corrected to average test day conditions by the following equations.

$$SHP_s = SHP_t \frac{(\delta_s \sqrt{\theta_s}) \left[\frac{N_R}{\sqrt{\theta}}\right]_s^3}{(\delta_t \sqrt{\theta_t}) \left[\frac{N_R}{\sqrt{\theta}}\right]_t^3} \tag{10}$$

$$V_{T_s} = V_{T_t} \frac{\left[\frac{N_R}{\sqrt{\theta}}\right]_s}{\left[\frac{N_R}{\sqrt{\theta}}\right]_t} \tag{11}$$

Where:

N_R = Main rotor speed (rev/min)

subscript t = Test day

subscript s = Average test day

Test data corrected for rate of climb, acceleration, instrumentation installation, and corrected to inherent sideslip, standard altitude, and ambient temperature are presented in figures 32 through 59, appendix E.

9. Level flight performance was determined by using equations 1 through 3, rewritten in the following form.

$$C_P = \frac{SHP(478935.3)}{\delta\sqrt{\theta} \left[\frac{N_R}{\sqrt{\theta}}\right]^3 \rho_o A R^3} \tag{12}$$

$$C_T = \frac{GW(91.19)}{\delta \left[\frac{N_R}{\sqrt{\theta}}\right]^2 \rho_o A R^2} \tag{13}$$

$$\mu = \frac{V_T(16.12)}{R\sqrt{\theta}\ \dfrac{N_R}{\sqrt{\theta}}} \tag{14}$$

Where:

478935.3 = Conversion factor (ft-lb-sec^2-rev^3/min^3-SHP)

91.19 = Conversion factor (sec^2-rev^2/min^2)

16.12 = Conversion factor (ft-rev/min-kt)

10. Data analysis was accomplished by plotting C_P versus μ for each test at the average C_T and $N_R/\sqrt{\theta}$. The curves through these data were then cross-faired as C_P versus C_T for lines of constant $N_R/\sqrt{\theta}$ at a given μ for an initial determination of what effect $N_R/\sqrt{\theta}$ had throughout the level flight envelope. These curves were subsequently faired into individual carpet plots (C_T versus C_P for lines of constant μ) at each $N_R/\sqrt{\theta}$ at the average test conditions (figs. 20 through 31, app E). The classification of these carpet plots into related families of curves (C_P versus $N_R/\sqrt{\theta}$ for lines of constant C_T at increments of μ) allows determination of power required as a function of airspeed for any value of C_T and $N_R/\sqrt{\theta}$ (figs. 4 through 19).

11. The specific range (SR) data were derived from the test level flight power required and fuel flow (W_{F_t}). Selected level flight performance SHP and fuel flow data for each engine were referred as follows.

$$SHP_{REF} = \frac{SHP_t}{\delta\theta^{0.5}} \tag{15}$$

$$W_{F_{REF}} = \frac{W_{F_t}}{\delta\theta^{0.55}} \tag{16}$$

A curve fit was subsequently spplied to this referred data and was used aa the basis to correct W_{F_t} to standard day fuel flow using the following equation.

$$W_{F_s} = W_{F_t} + \Delta W_F \tag{17}$$

Where:

ΔW_F = Change in fuel flow between SHP_t and SHP_s

The following equation was used for determination of SR.

$$SR = \frac{V_{T_s}}{W_{F_s}} \tag{18}$$

Stabilator Position Effect:

12. Tests were flown in ball-centered level flight at a predetermined C_T and μ. Stabilator position was varied incrementally up and down from the trim schedule position to a predetermined limit based on the main rotor mast endurance limit. Change in power required for level flight due to chsnge in stabilator position for a constant μ was obtained at each stabilized increment. Power corrections indentical to those used in the level flight performance analysis, equation 9, were also applied. Plotting atabilator movement and corresponding change in power required show they vary as a function of μ and C_T (figs. 69 and 70, app E). Direction of stabilator movement indicates if the change in power required is additive or subtractive.

13. Stabilator position is a function of collective position and indicated airspeed in stabilized ball-centered level flight. Different dimensional conditions, and correapondingly different stabilator positions, can produce the same nondimensional condition. Collective position analyzed on a nondimensional basis normalizes as a function of μ for the same C_T regardless of the dimensional circumstances. Indicated airspeed varies with dimensional conditions for the same μ. Level flight power required, therefore, can be adjusted for the effects of different stabilator positions caused by flying at different test conditions for the same C_T. The procedure is to determine the difference in indicated airspeed for the same μ and convert this difference into change of stabilator position

and consequently ΔC_P. Data at the different test conditions allows solving equation 14 for V_T, and determining δ as follows.

$$\sigma = [1-6.8755856E-06\ (H_D)]^{4.25584} \tag{19}$$

Where:

H_D = Density altitude (ft)

$$\delta = \sigma\theta \tag{20}$$

Calibrated airspeed (V_{cal}) and consequently indicated airspeed (V_{ic}) at the different test conditions are determined as follows.

$$V_{cal} = 1479.12 \left(\left\{ \left[\delta \right] \left\{ \left[1 + 0.2 \left(\frac{V_T}{38.97(OAT+273.15)^{1/2}} \right)^2 \right]^{7/2} -1 \right\} +1 \right\}^{2/7} -1 \right)^{1/2} \tag{21}$$

V_{ic} = f (V_{cal}, Ship airspeed system position error, fig. 71, app E)

The difference in stabilator position between test conditions can be obtained from the slopes of the airspeed versus stabilator angle schedule.

$$\Delta\ STAB = \Delta V_{ic} \left[\frac{\Delta\ STAB}{\Delta V_{ic}} \right]_{segment\ 1} + \Delta V_{ic} \left[\frac{\Delta\ STAB}{\Delta V_{ic}} \right]_{segment\ 2} + \ldots$$

Where:

Δ STAB = Difference in stabilator position (deg)

ΔV_{ic} = Difference in indicated airspeed within an airspeed segment (kt)

$$\frac{\Delta \text{ STAB}}{\Delta V_{ic}}$$ = Slope of stabilator schedule within an airspeed segment (deg/kt)

0 deg/kt;	above 147 KIAS
0.1042 deg/kt;	80 to 147 KIAS
0.7250 deg/kt;	35 to 80 KIAS
0 deg/kt;	below 35 KIAS

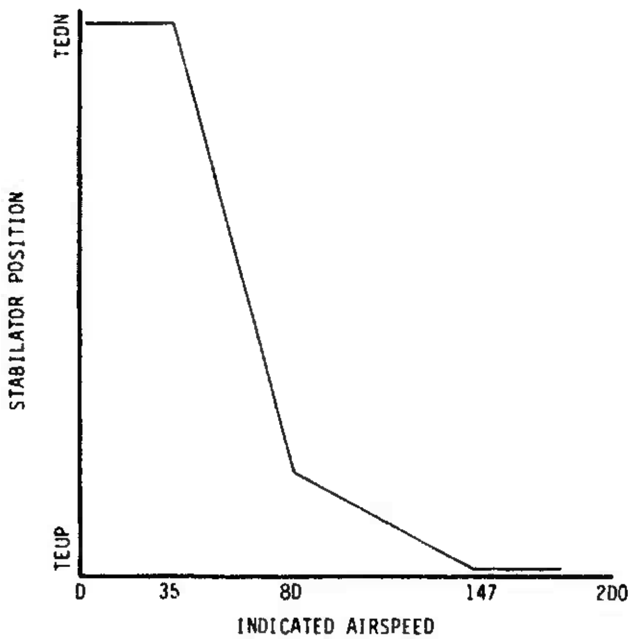


Figure 1. Normalized Stabilator Schedule

The change in power required to correct for differences in stabilator position is obtained when curves from figure 68, appendix E, are cross-faired as ΔC_P versus Δstabilator angle for a specific μ.

$$C_{P(\text{test condition 2})} = C_{P(\text{test condition 1})} \pm \Delta C_P$$

Where:

+ or - is employed depending on direction of stabilator movement when transversing from test condition 1 to test condition 2.

+ ;TEUP movement
- ;TEDN movement

TEUP = Stabilator trailing edge up
TEDN = Stabilator trailing edge down

APPENDIX E. TEST DATA

INDEX

FIGURE 1
SUMMARY HOVER PERFORMANCE
UH-60A USA S/N 82-23748
MAIN ROTOR SPEED = 258 RPM

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
3. WINDS LESS THAN THREE KNOTS
4. HOVER DATA DERIVED FROM FIGURES 2 AND 3
5. INTERMEDIATE (30 MINUTE LIMIT) POWER AVAILABLE OBTAINED FROM USAAEFA REPORT NO. 77-17
6. SOLID LINES DENOTE 100 FOOT MAIN WHEEL HEIGHT
7. DASHED LINES DENOTE SINGLE ENGINE OPERATION AT A 2 FOOT MAIN WHEEL HEIGHT

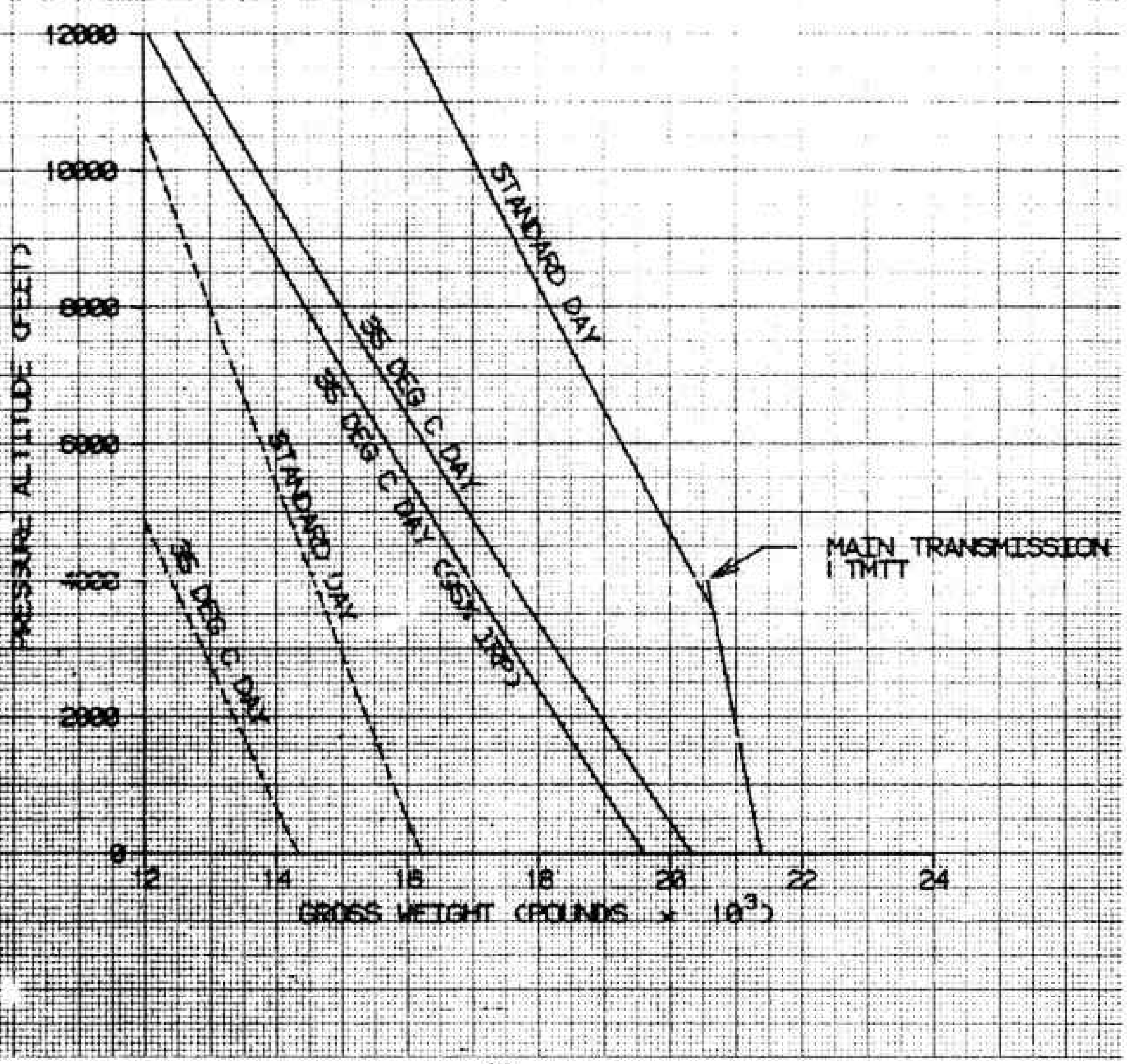


FIGURE 2
NONDIMENSIONAL HOVER PERFORMANCE
UH-60A USA S/N 82-23748
WHEEL HEIGHT = 2 FT
NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)

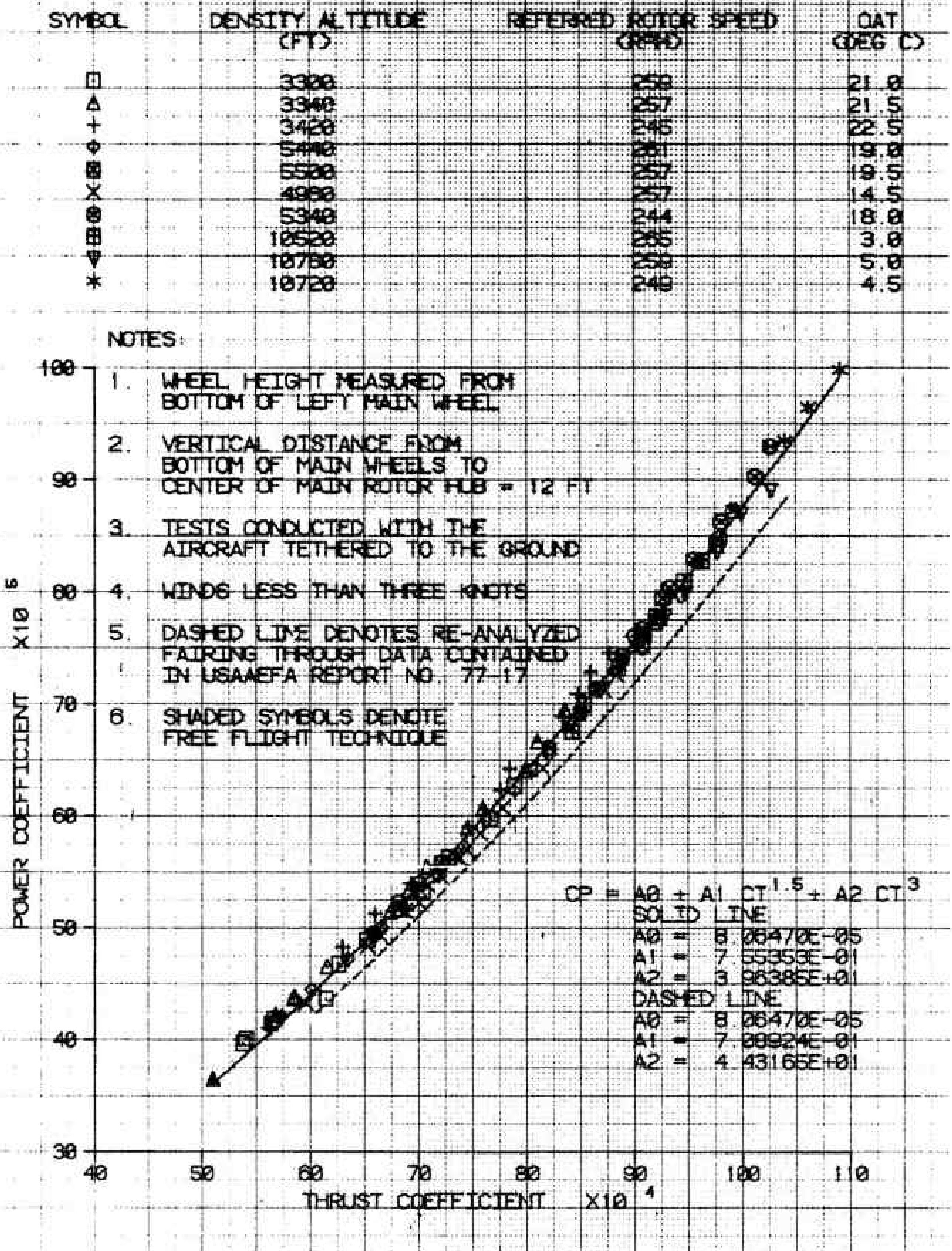


FIGURE 3
NONDIMENSIONAL HOVER PERFORMANCE
UH-60A USA S/N 82-23748
WHEEL HEIGHT = 100 FT
NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)

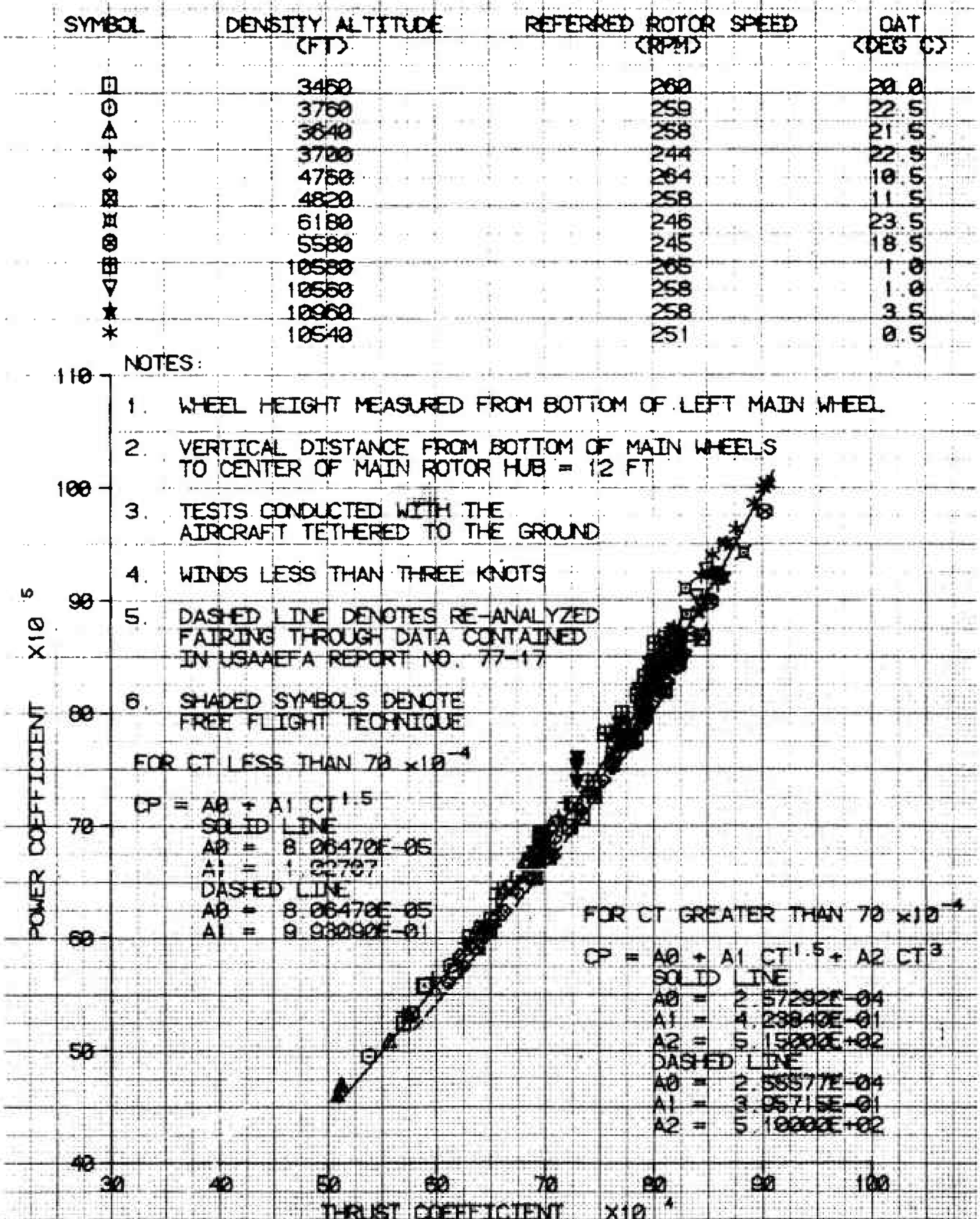


FIGURE 4

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

μ = 0.10

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

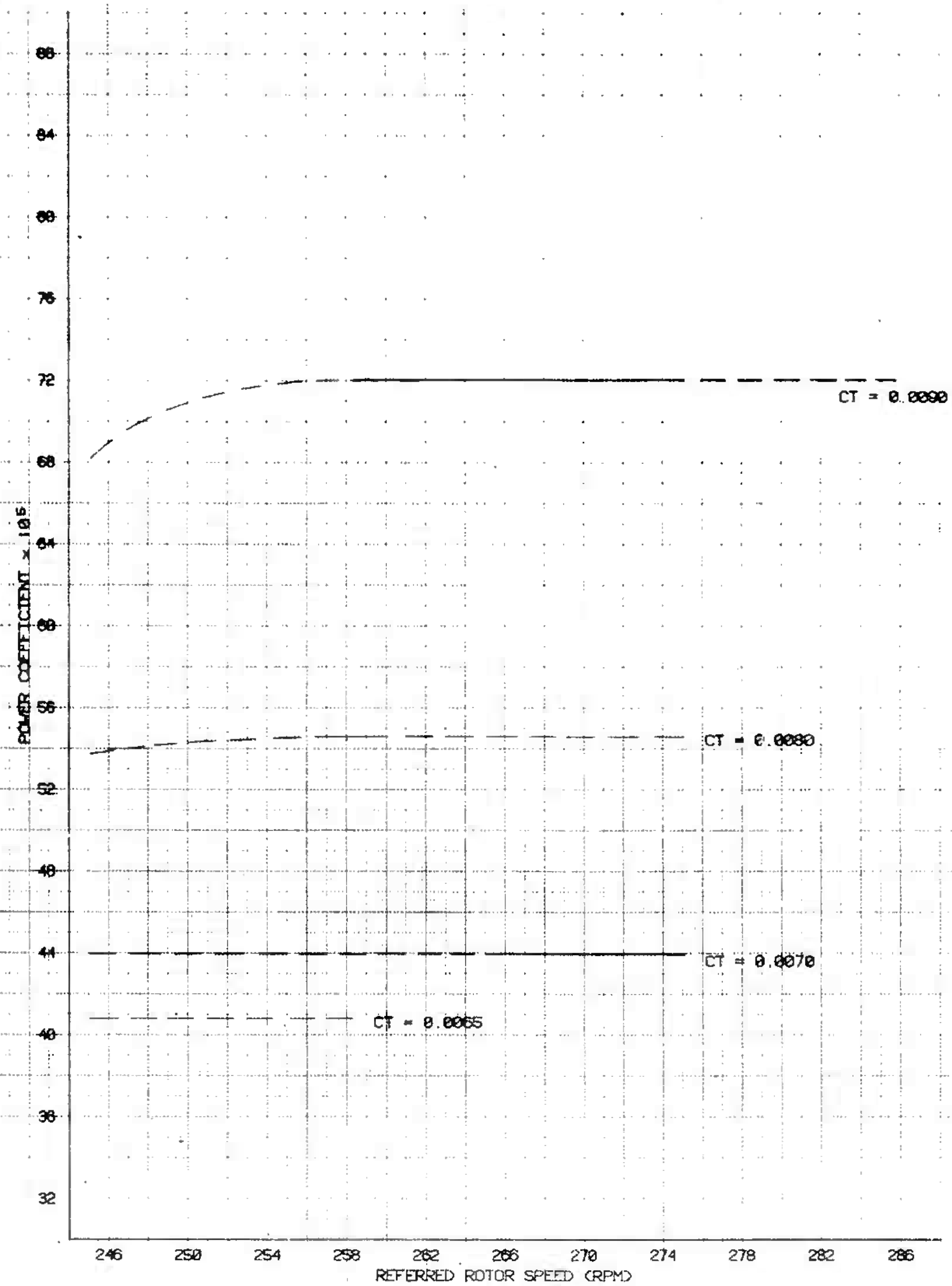


FIGURE 5
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

μ = 0.12

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

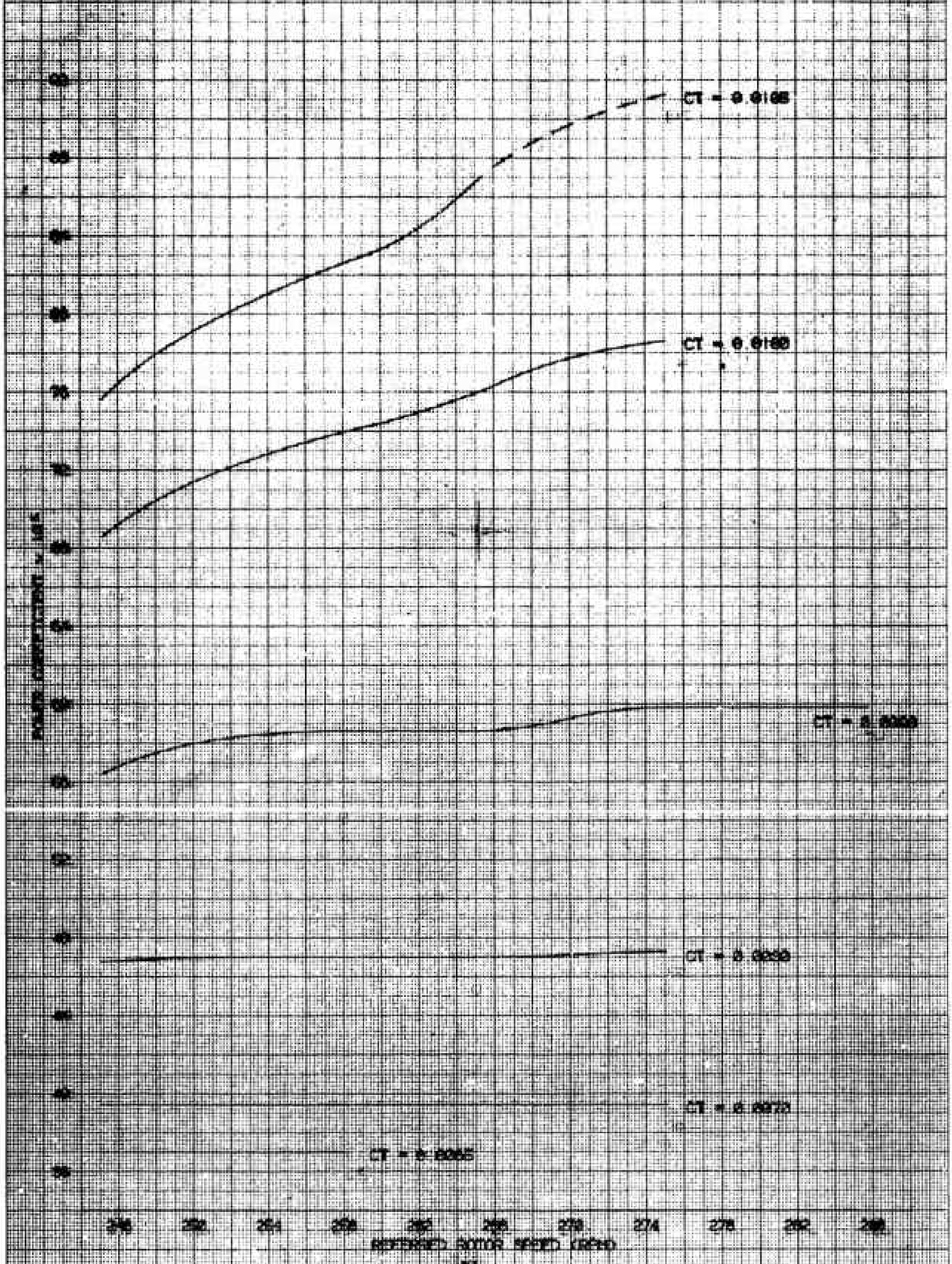


FIGURE 6

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.14$

NOTES

1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

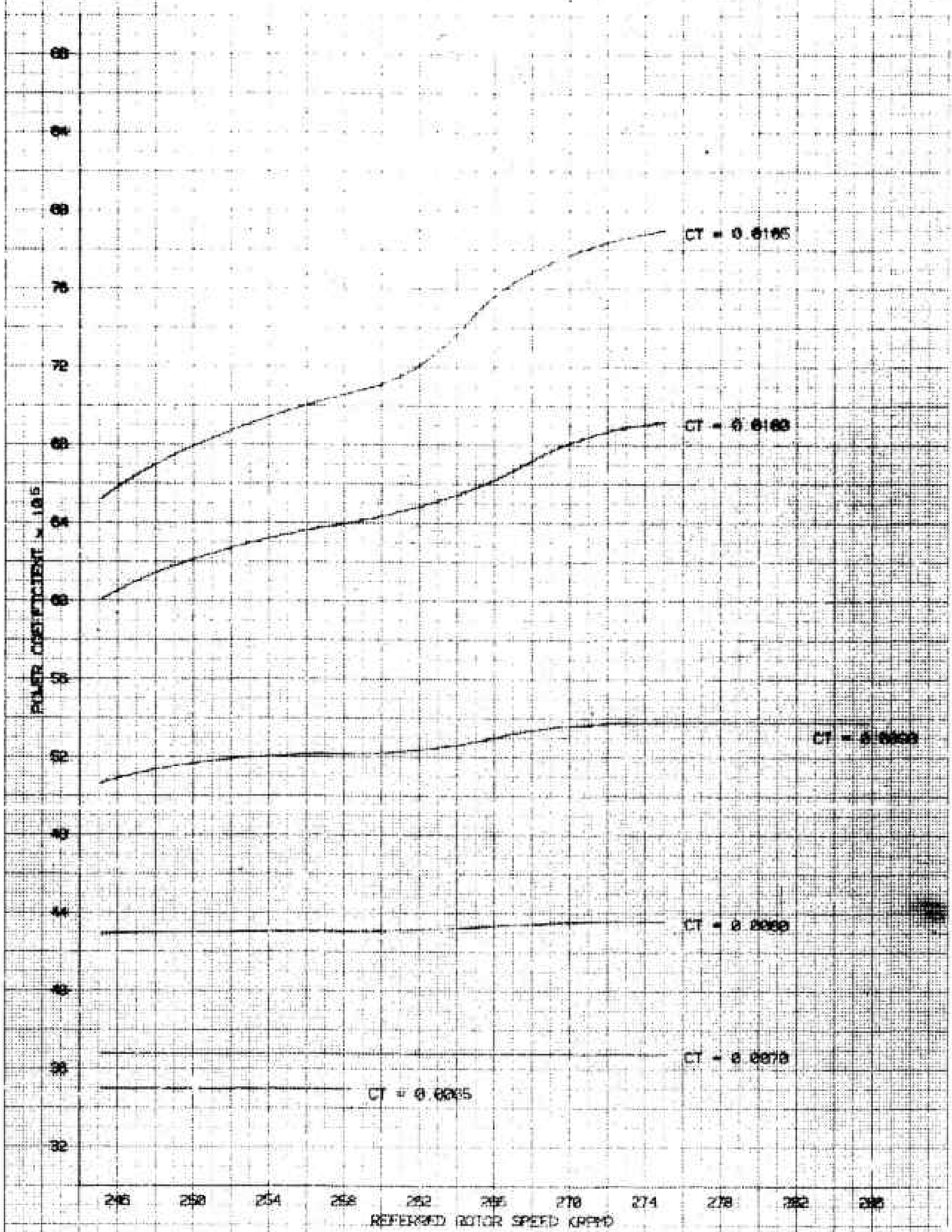


FIGURE 7
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.16$

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

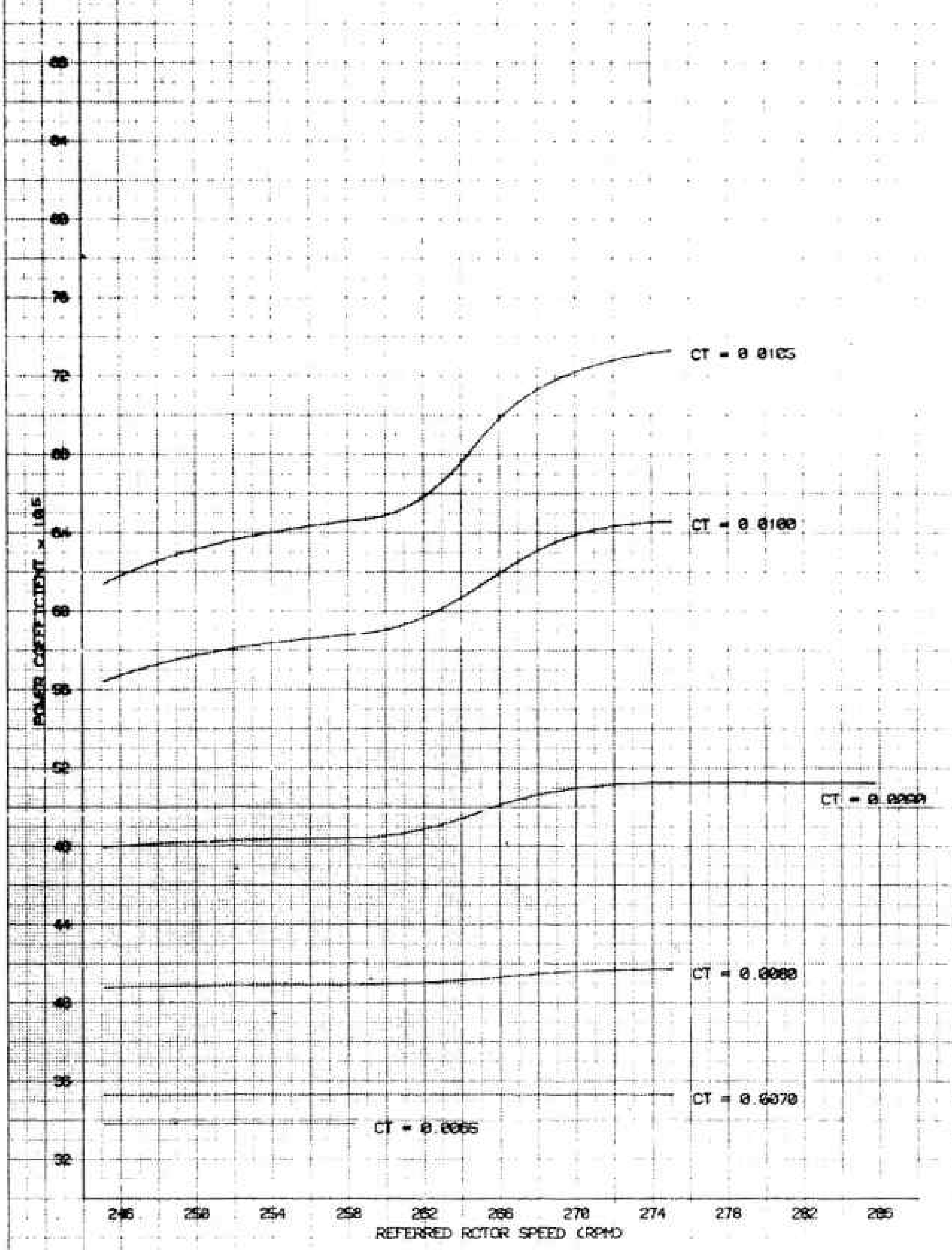


FIGURE 8

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.18$

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

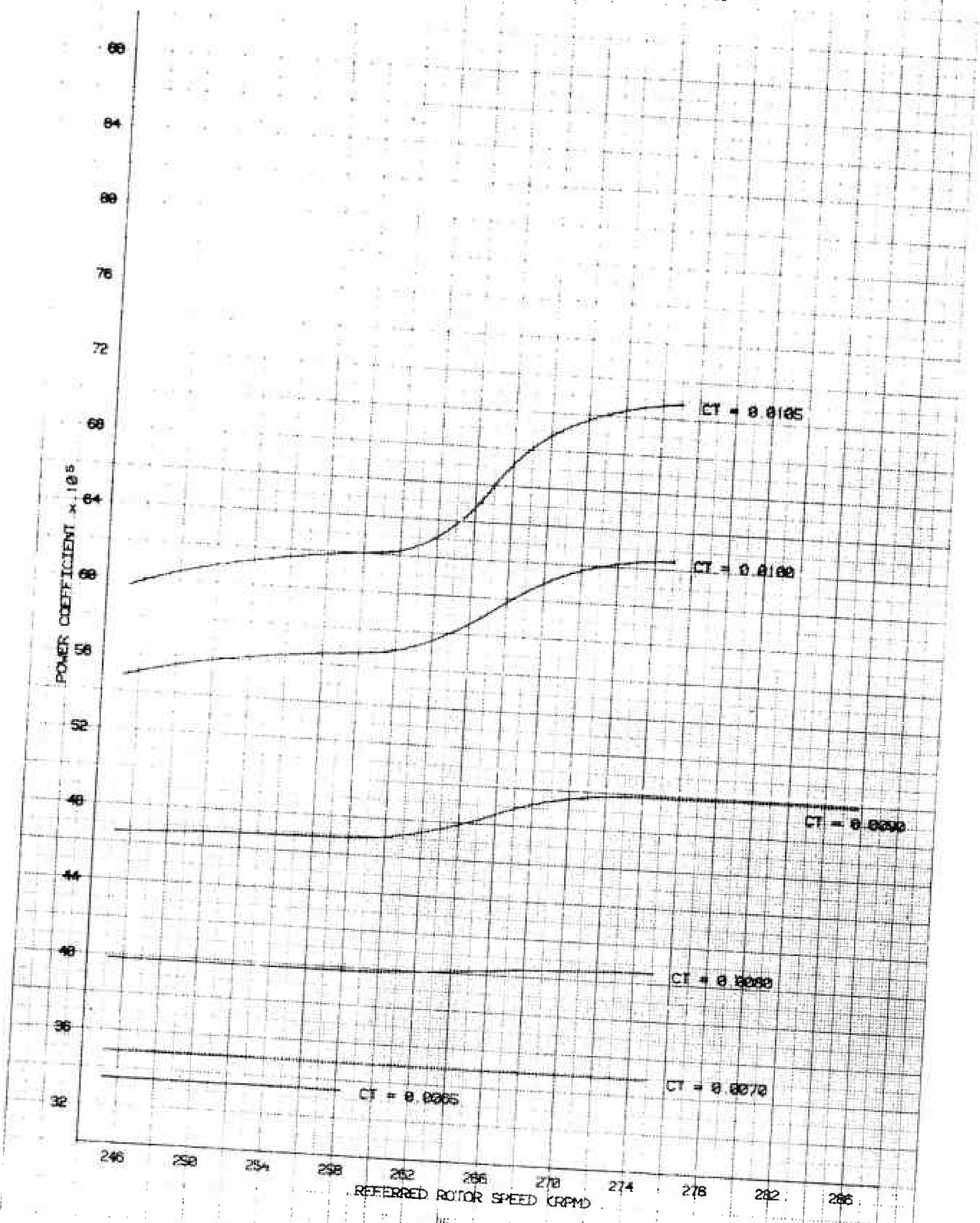


FIGURE 9
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

μ = 0.20

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

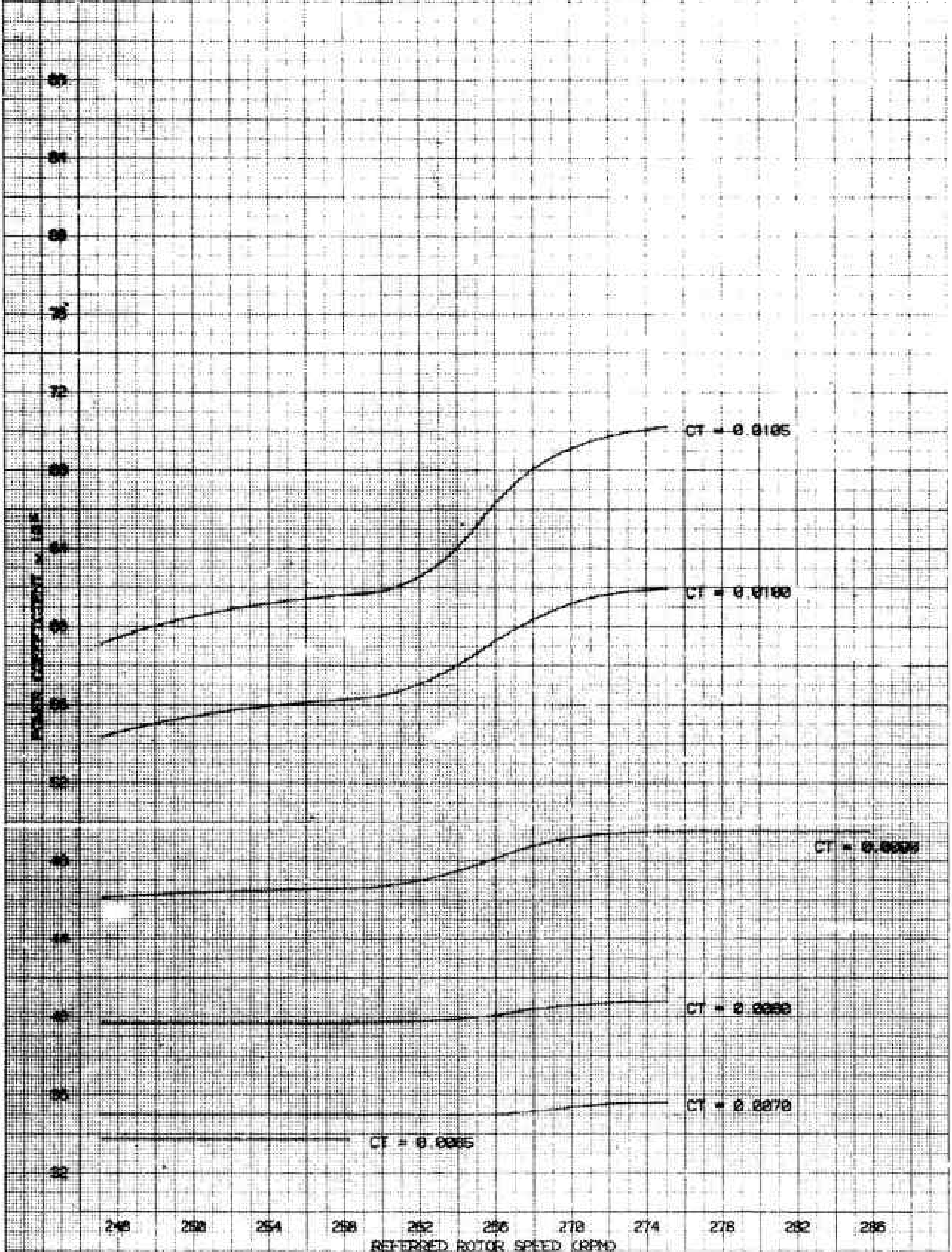


FIGURE 10
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.22$

NOTES
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

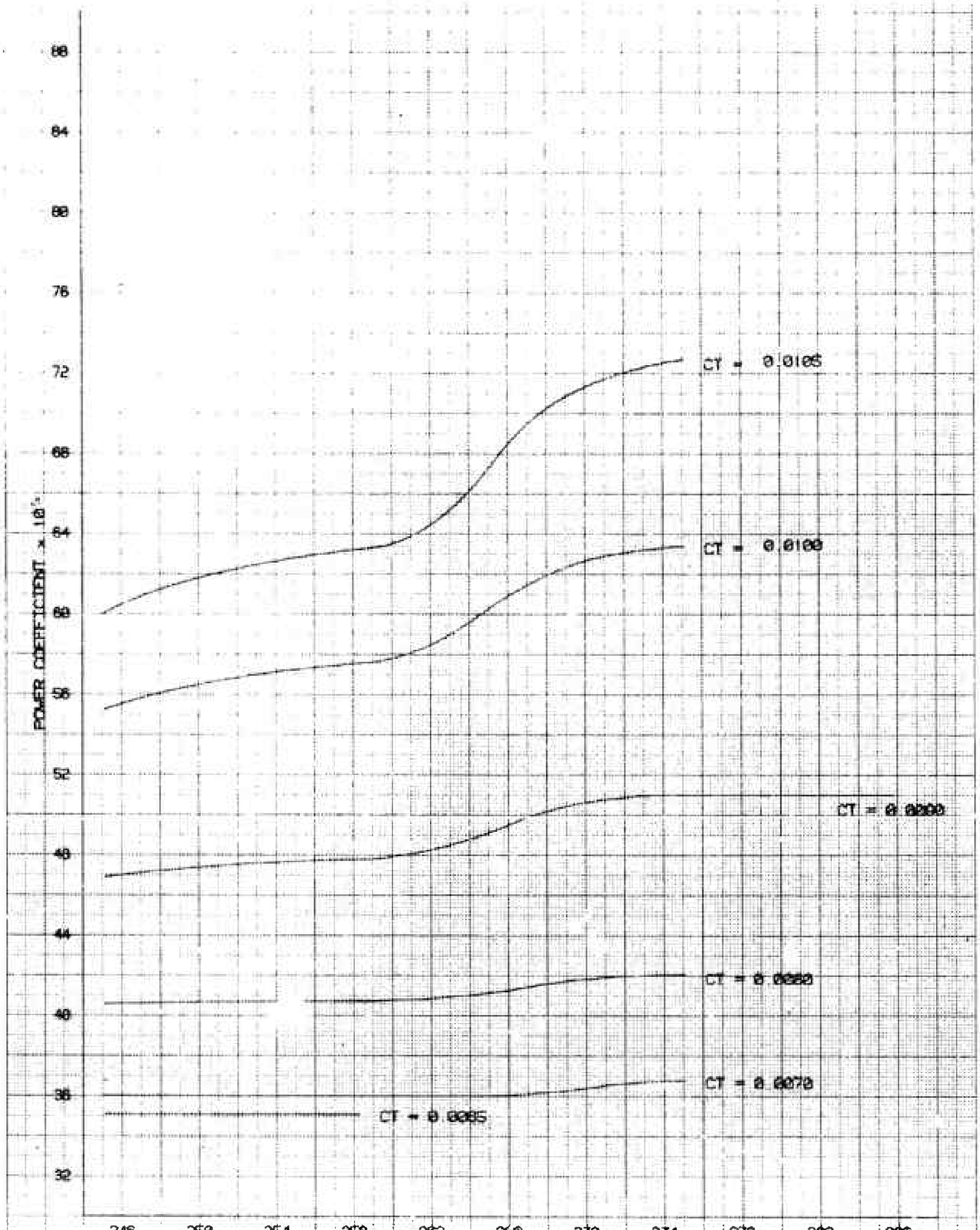


FIGURE 11
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

μ = 0.24

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

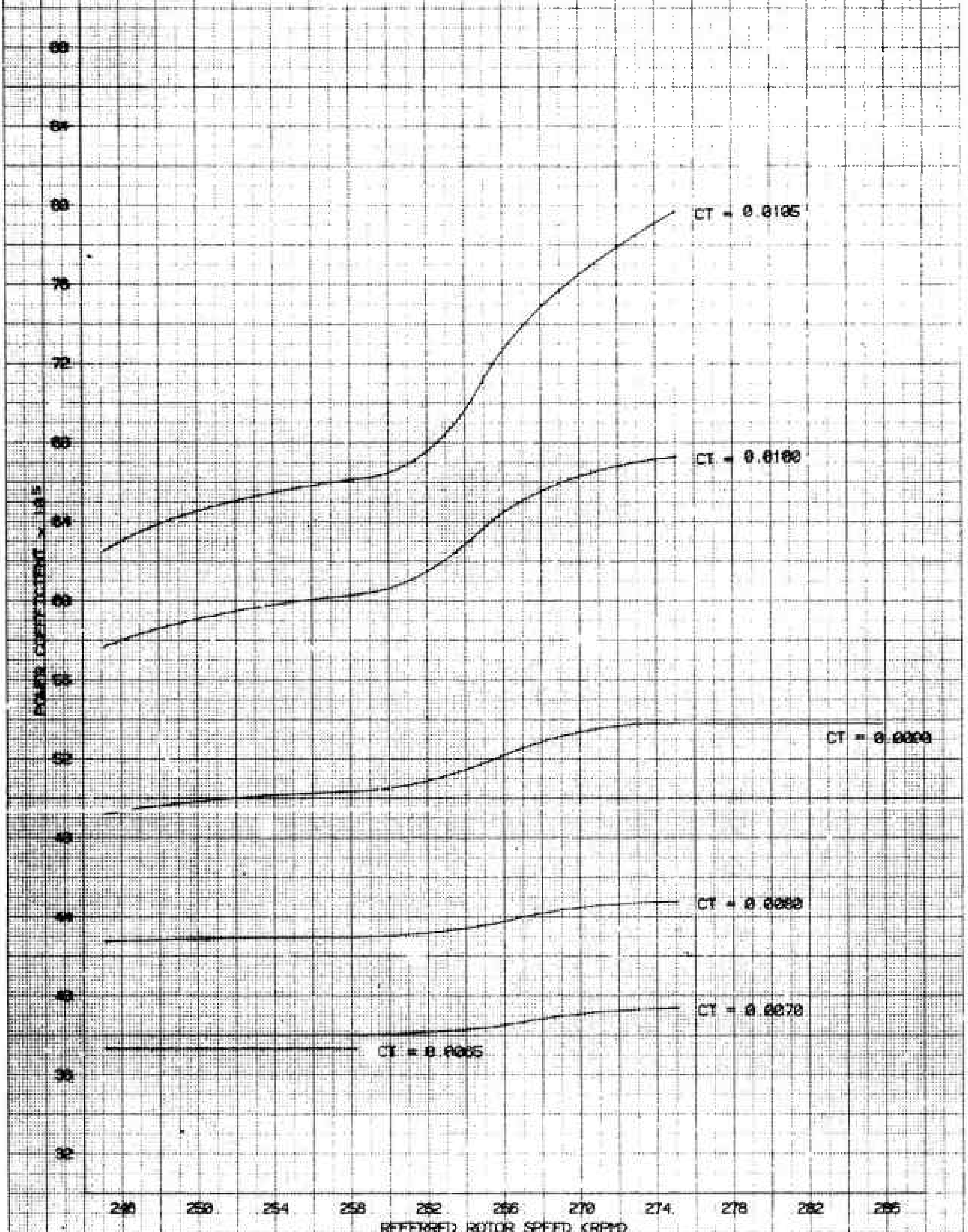


FIGURE 12

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.26$

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

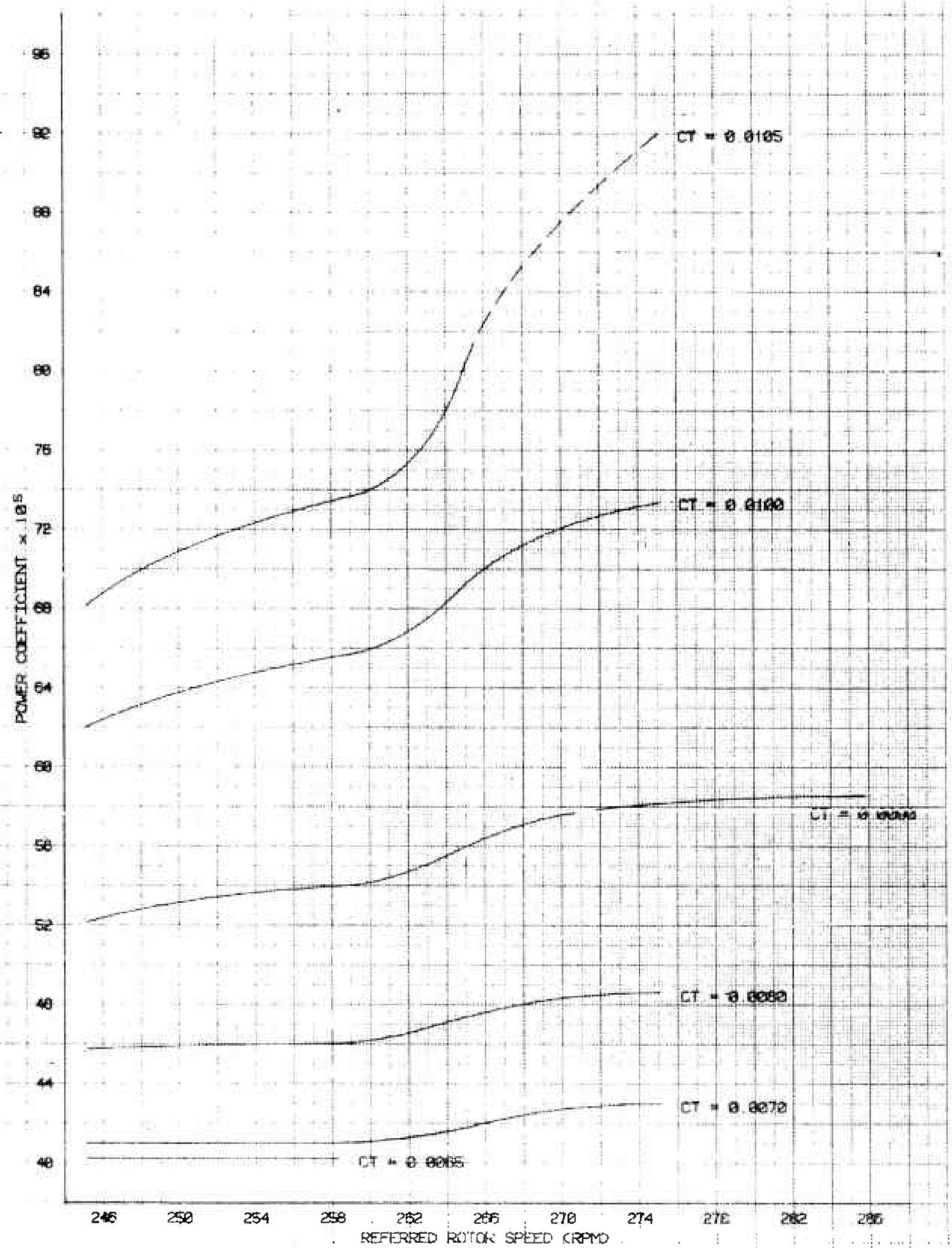


FIGURE 13
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.20$

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

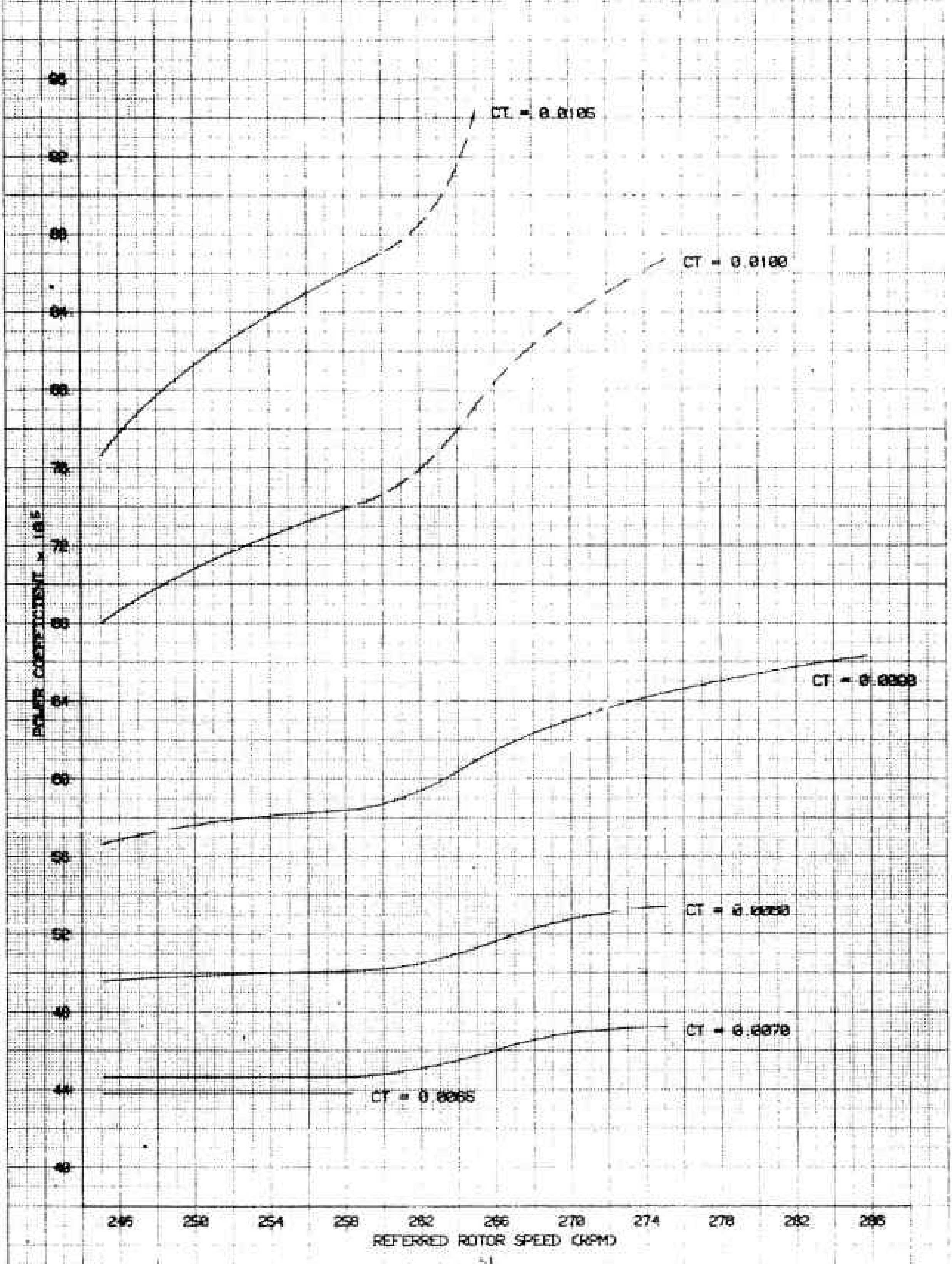


FIGURE 14
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.30$

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31 AND 53

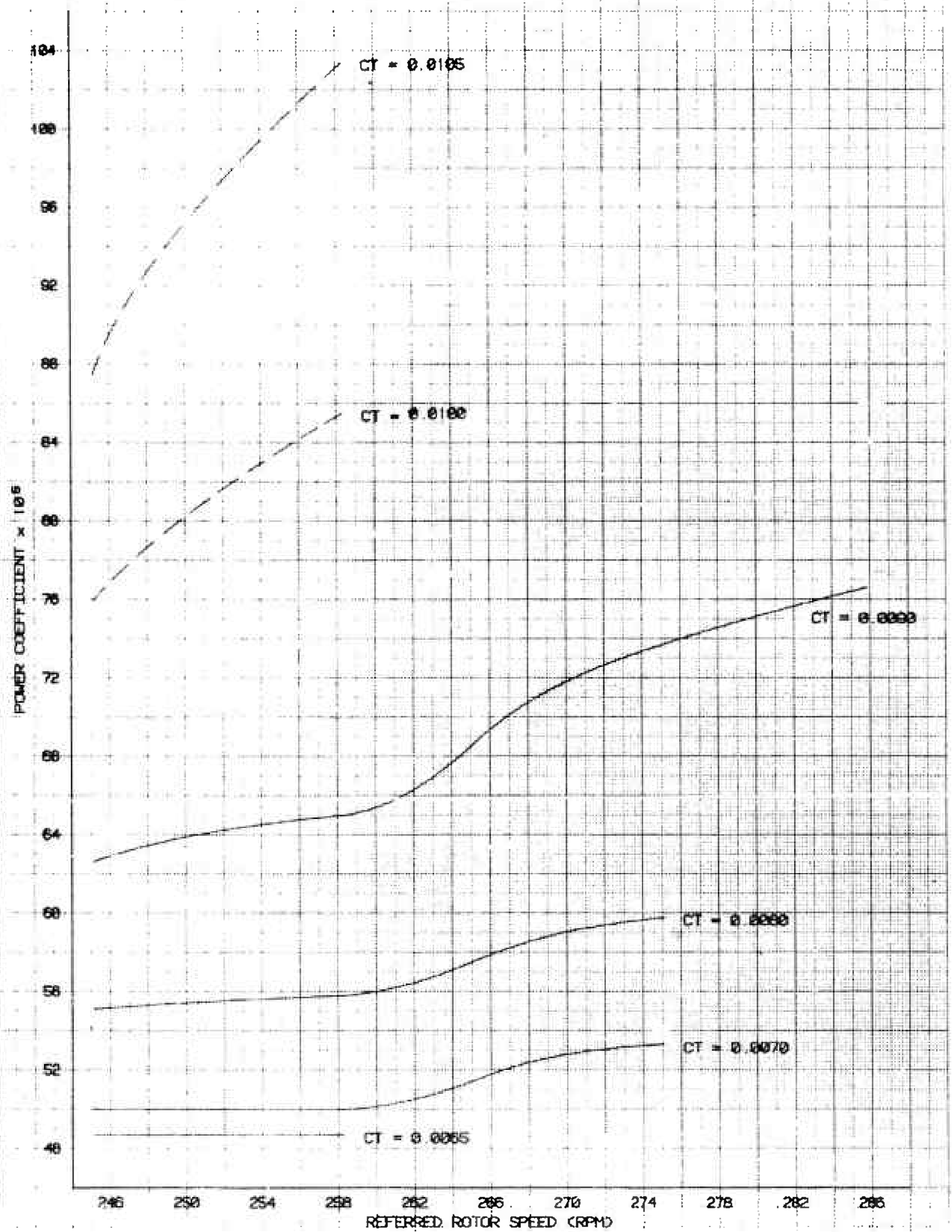


FIGURE 15

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.32$

NOTES

1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31

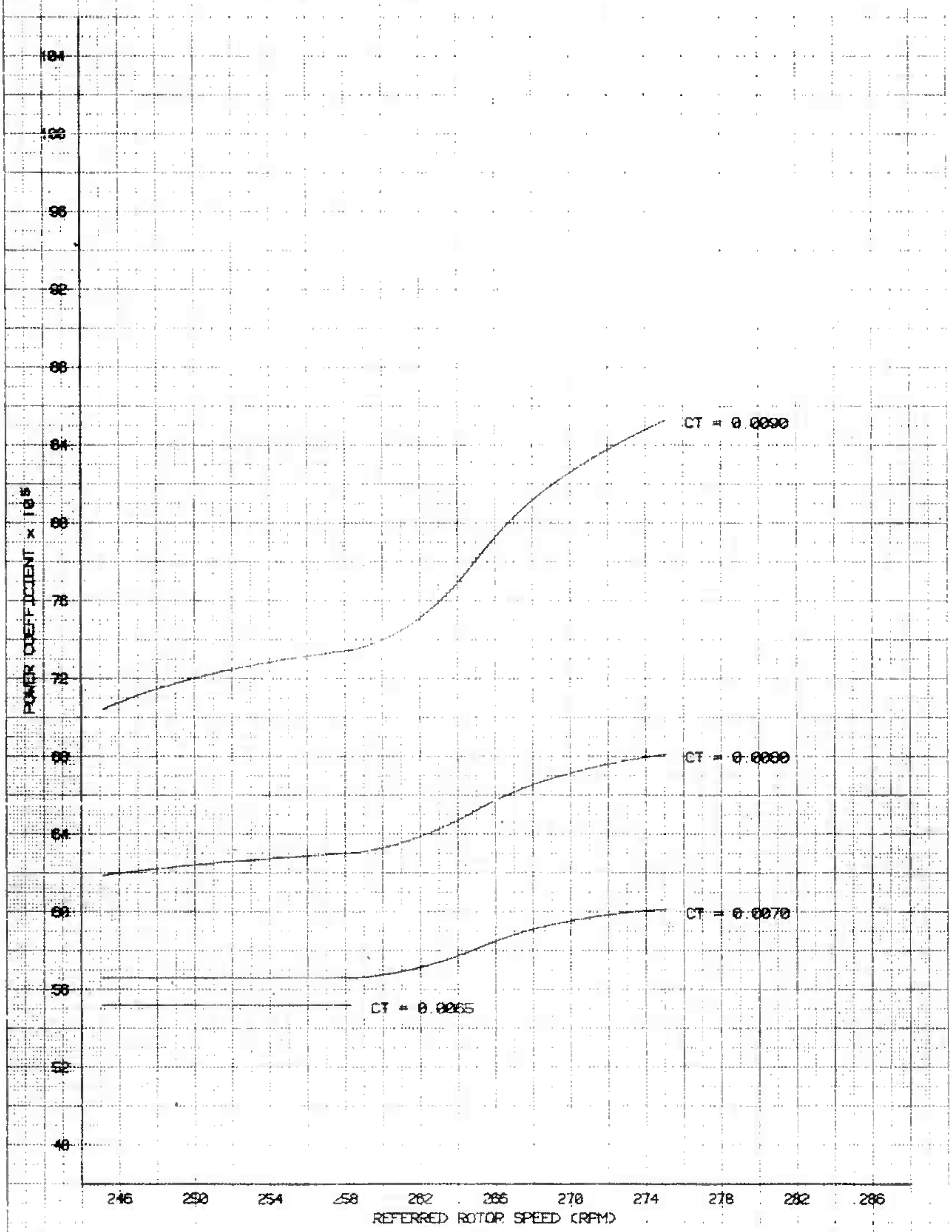


FIGURE 16
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

$\mu = 0.34$

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31

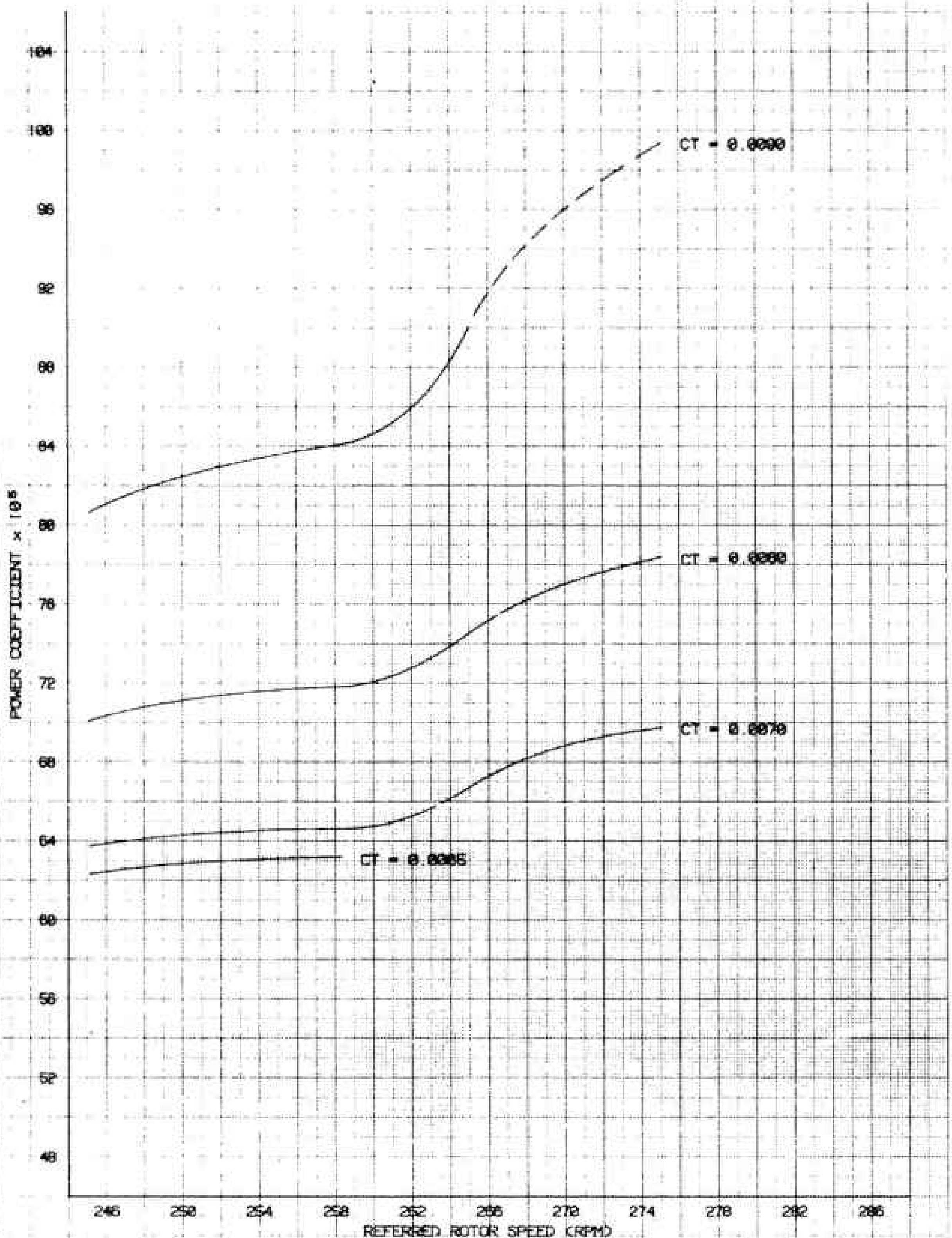


FIGURE 17
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

μ = 0.36

NOTES
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31

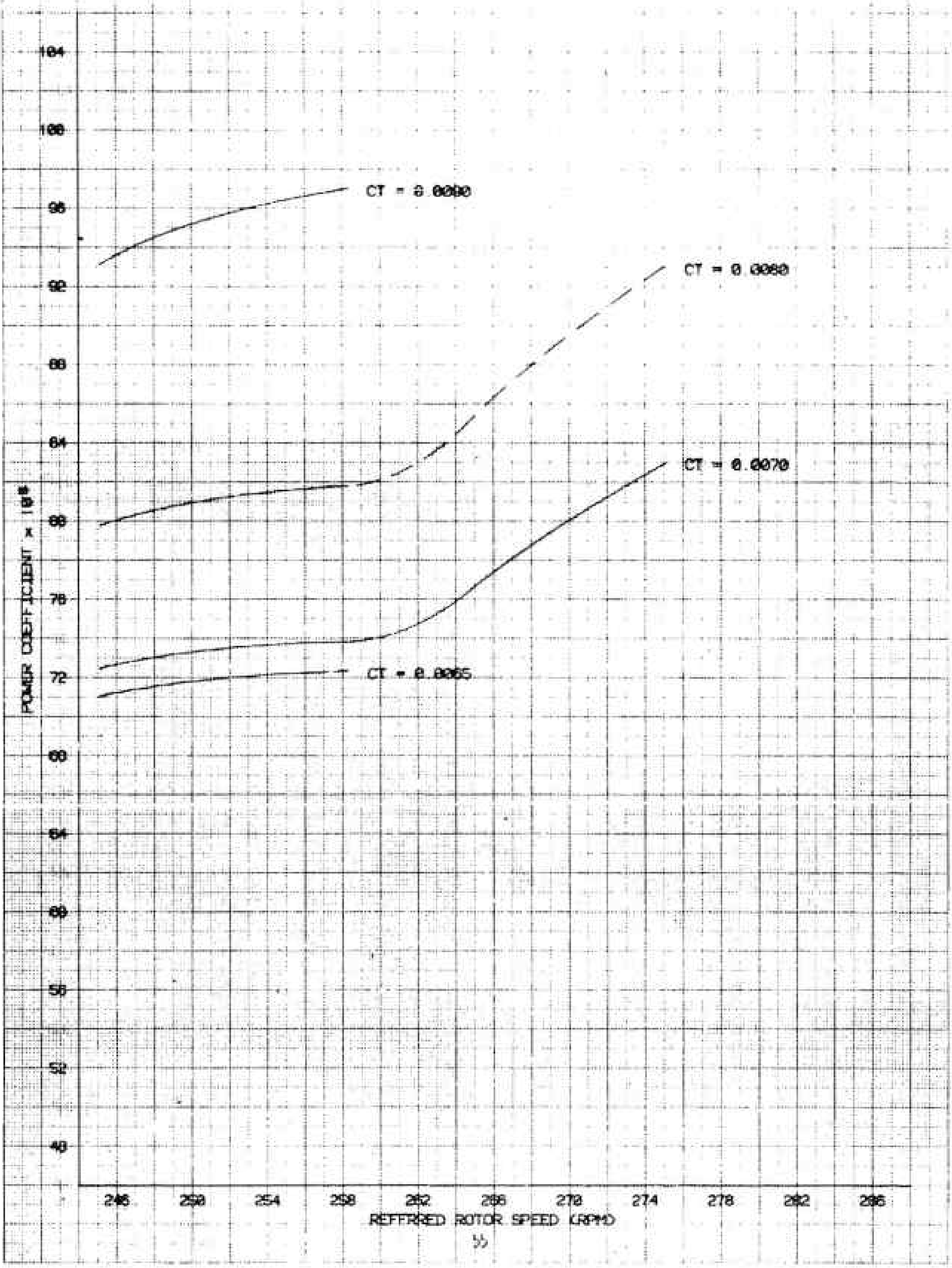


FIGURE 18
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716 and S/N 82-23748

μ = 0.30

NOTES
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 31

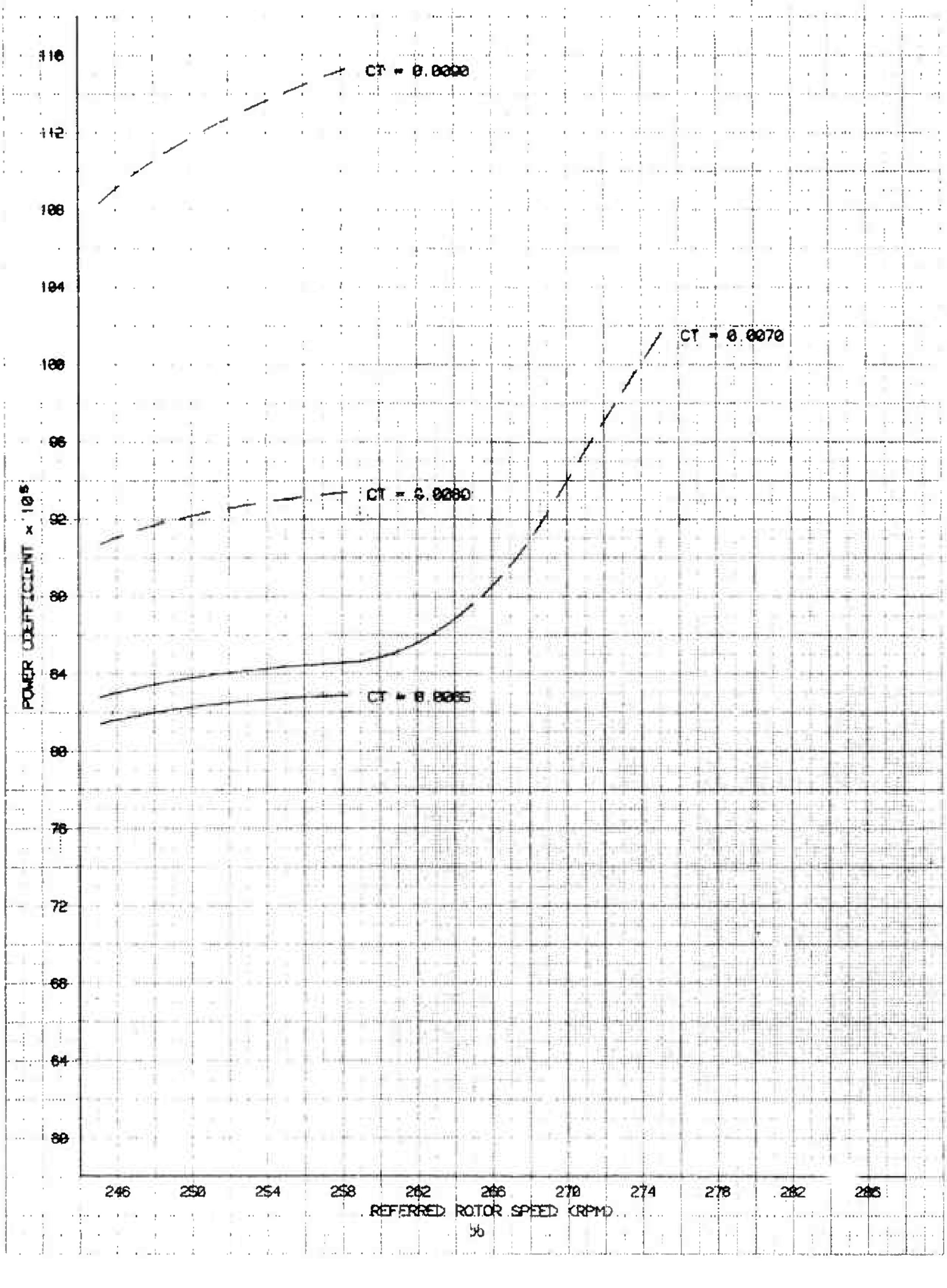


FIGURE 19

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

μ = 0.40

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. CURVES OBTAINED FROM FIGURES 20 THRU 28

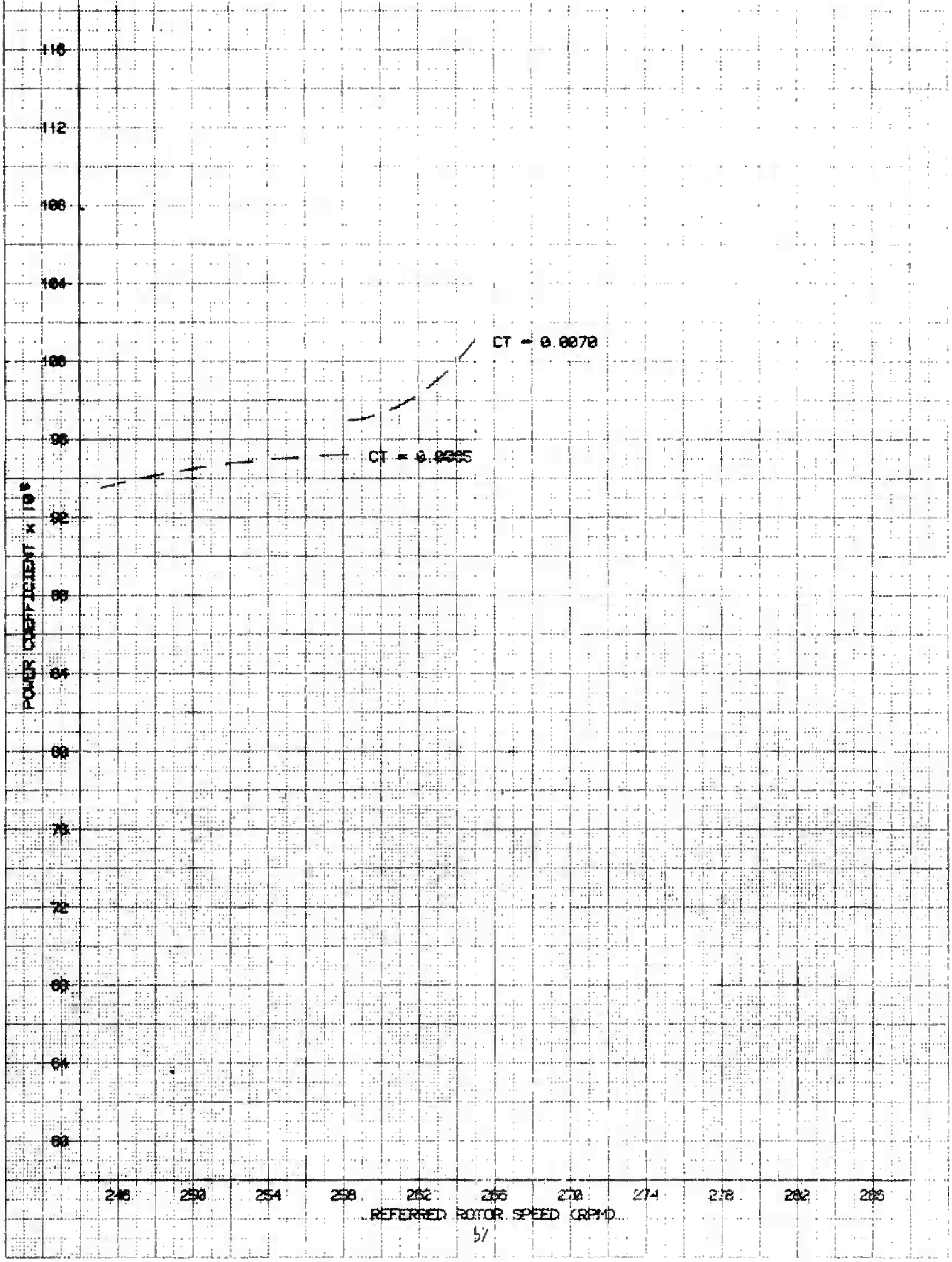


FIGURE 20
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 245.1

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.1
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.3 LEFT
5. POINTS DERIVED FROM FIGURES 32 THRU 34

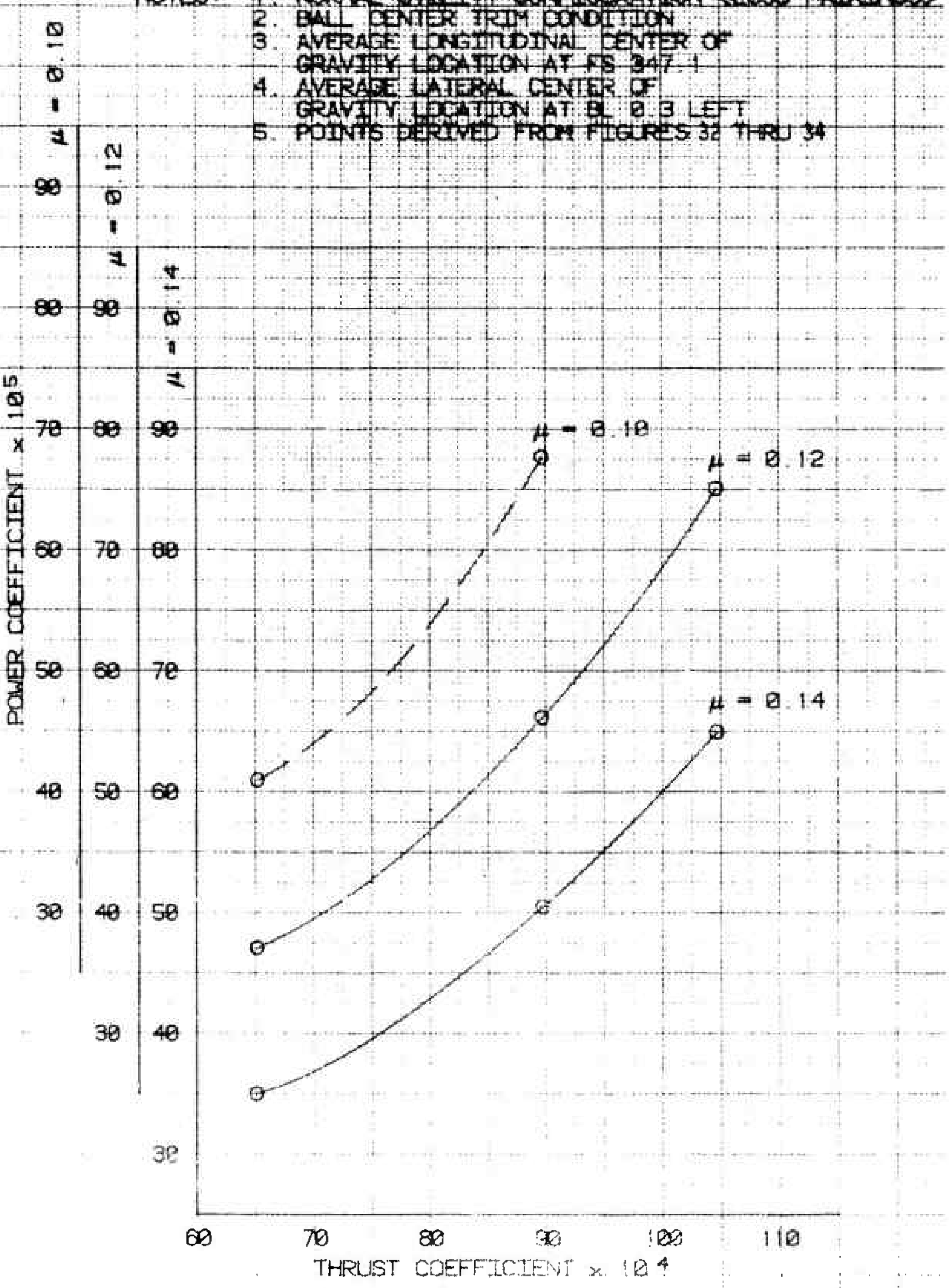


FIGURE 21

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 245.1

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.1
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.3 LEFT
5. POINTS DERIVED FROM FIGURES 32 THRU 34

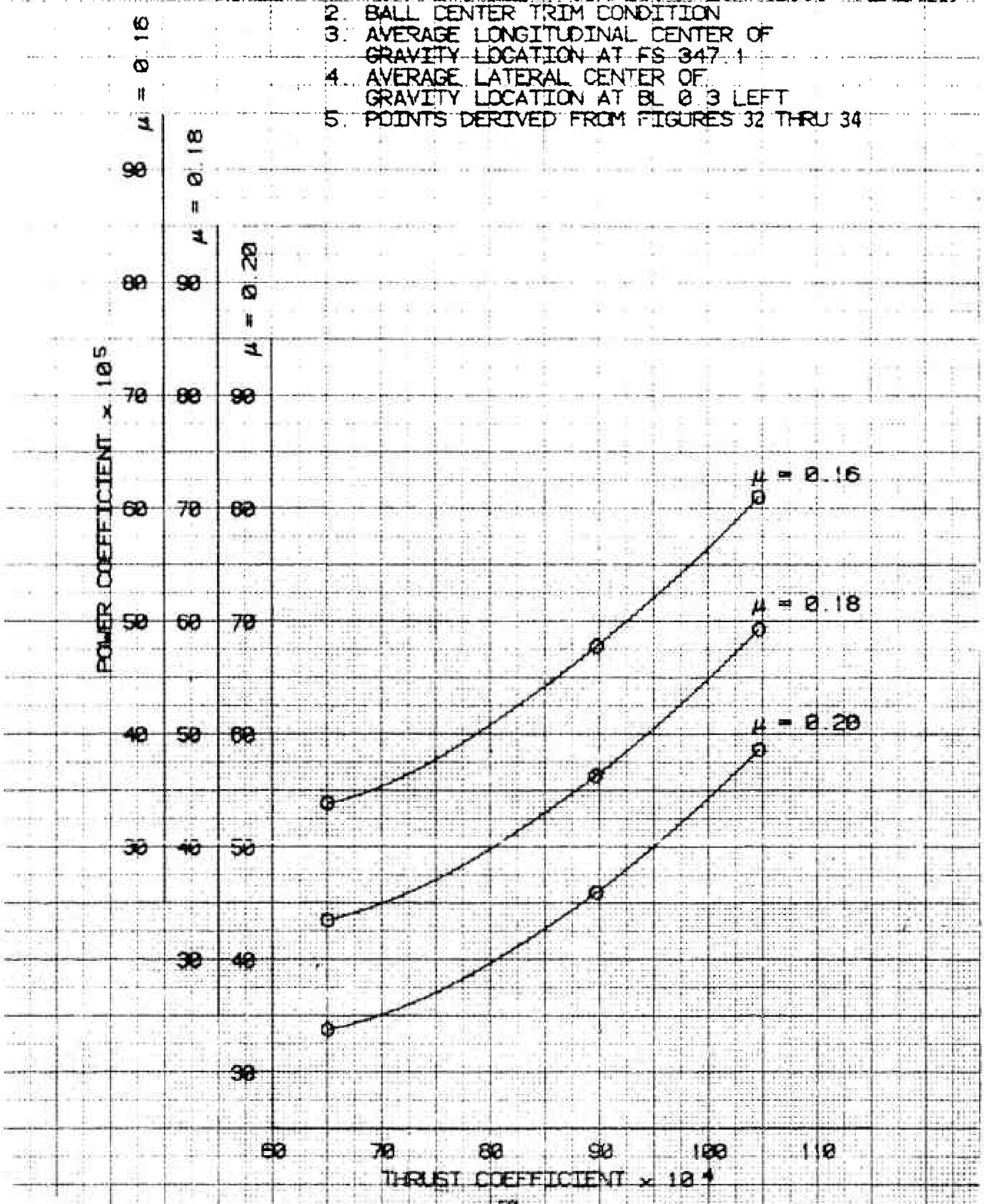


FIGURE 22

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 245.1

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.1
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.3 LEFT
5. POINTS DERIVED FROM FIGURES 32 THRU 34

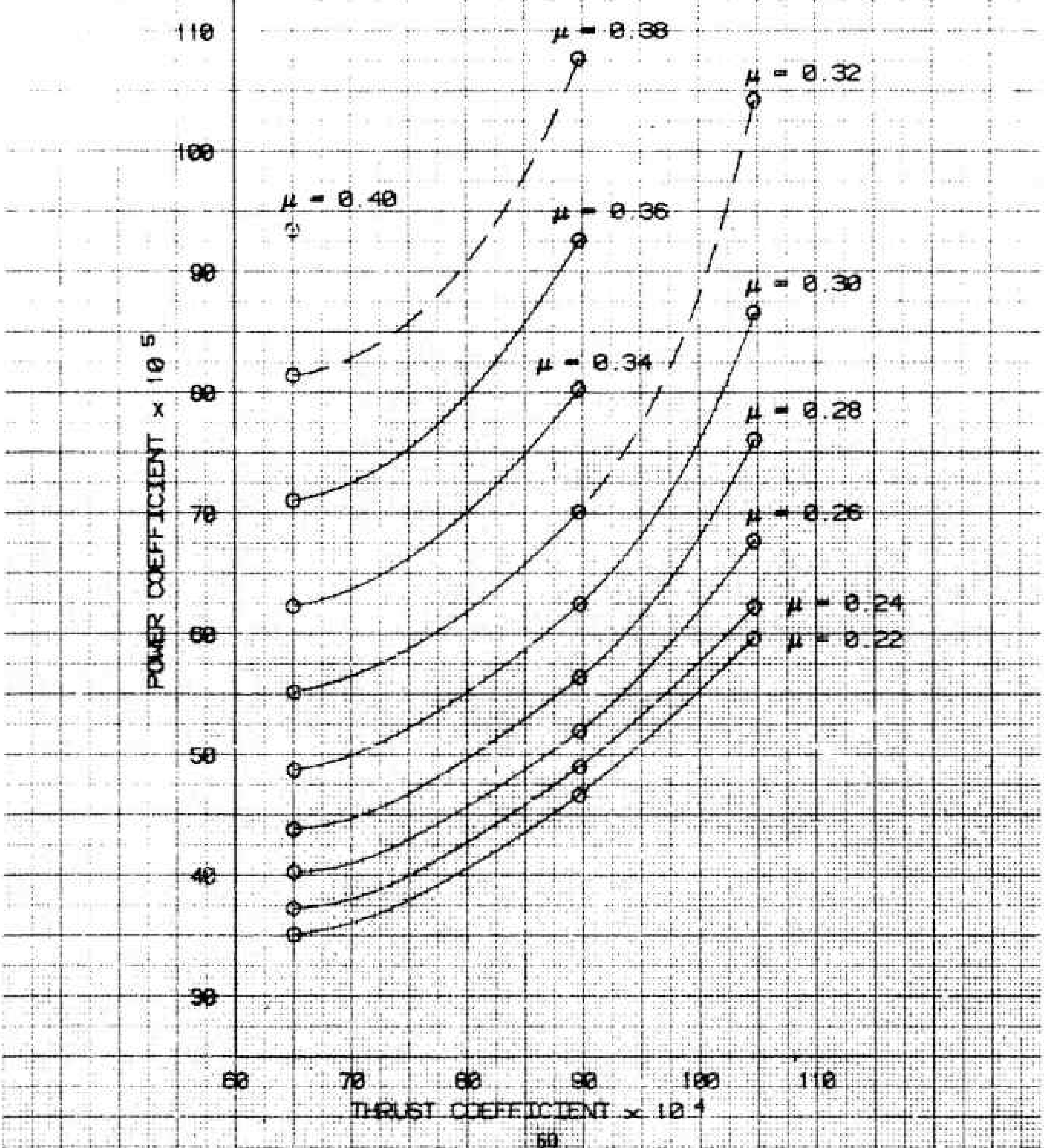


FIGURE 23

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

REFERRED ROTOR SPEED = 258.3

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. POINTS DERIVED FROM FIGURES 35 THRU 44

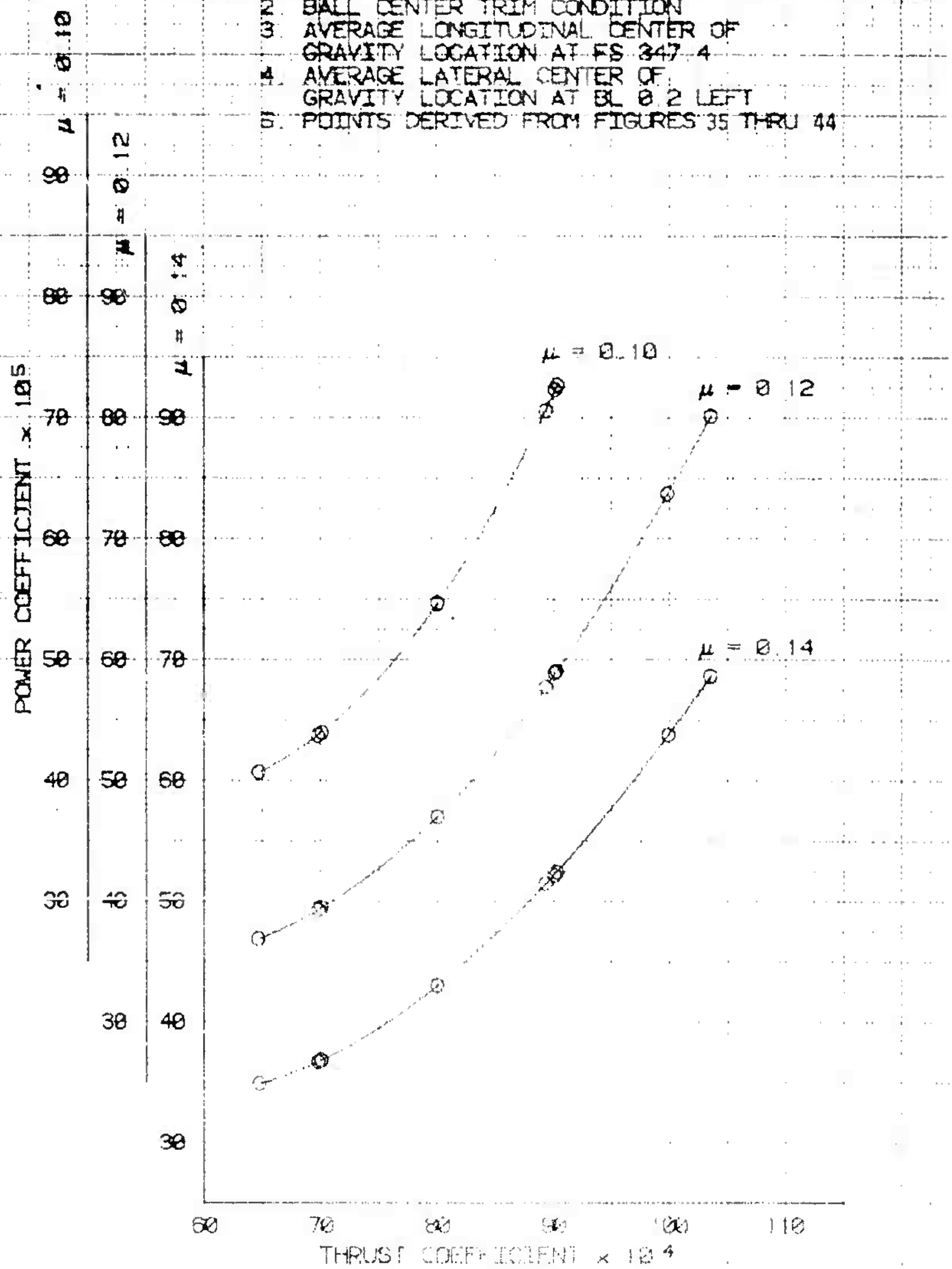


FIGURE 24

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

REFERRED ROTOR SPEED = 258.3

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. POINTS DERIVED FROM FIGURES 35 THRU 44

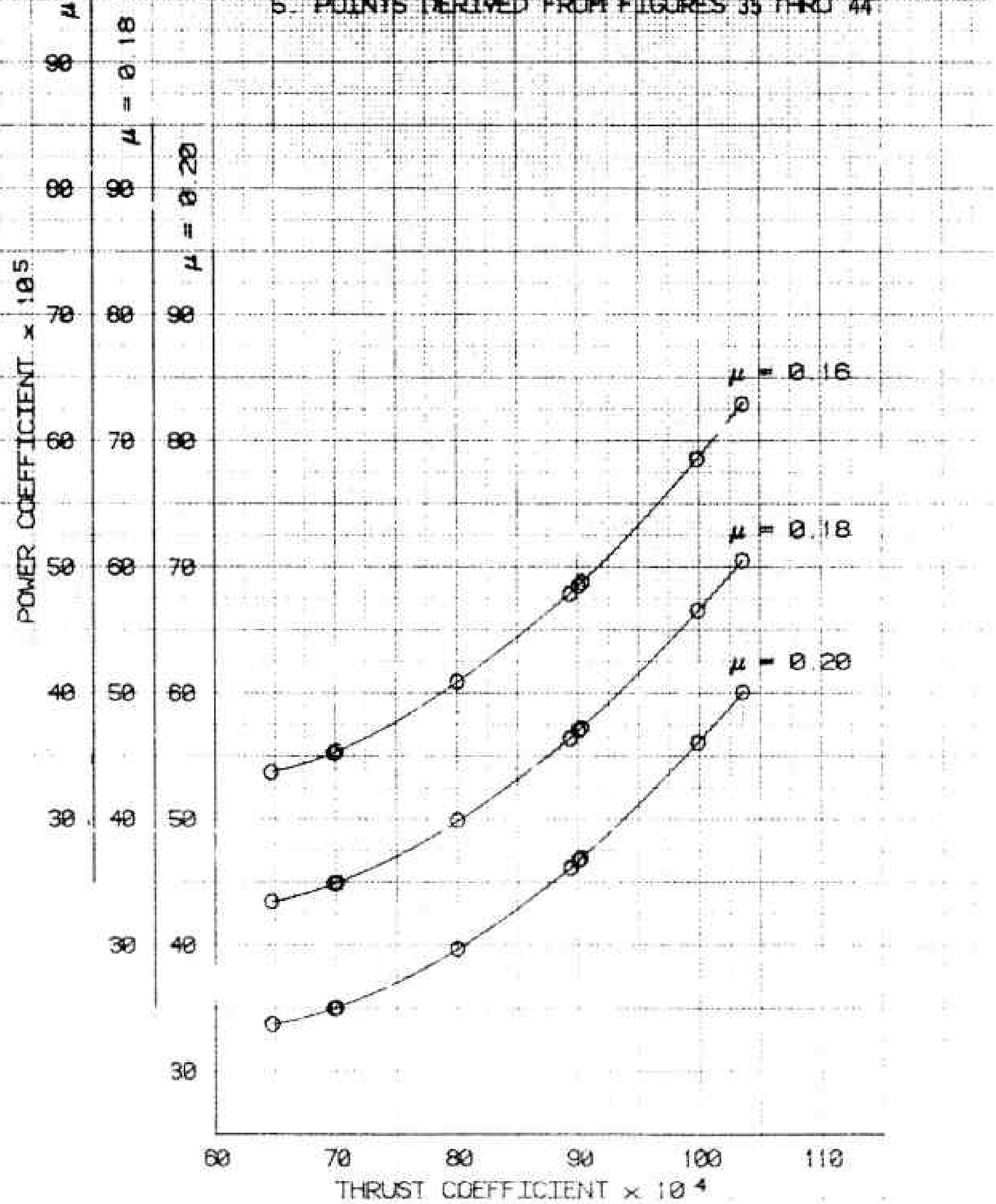


FIGURE 25

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716 and S/N 82-23748

REFERRED ROTOR SPEED = 258.3

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. POINTS DERIVED FROM FIGURES 35 THRU 44

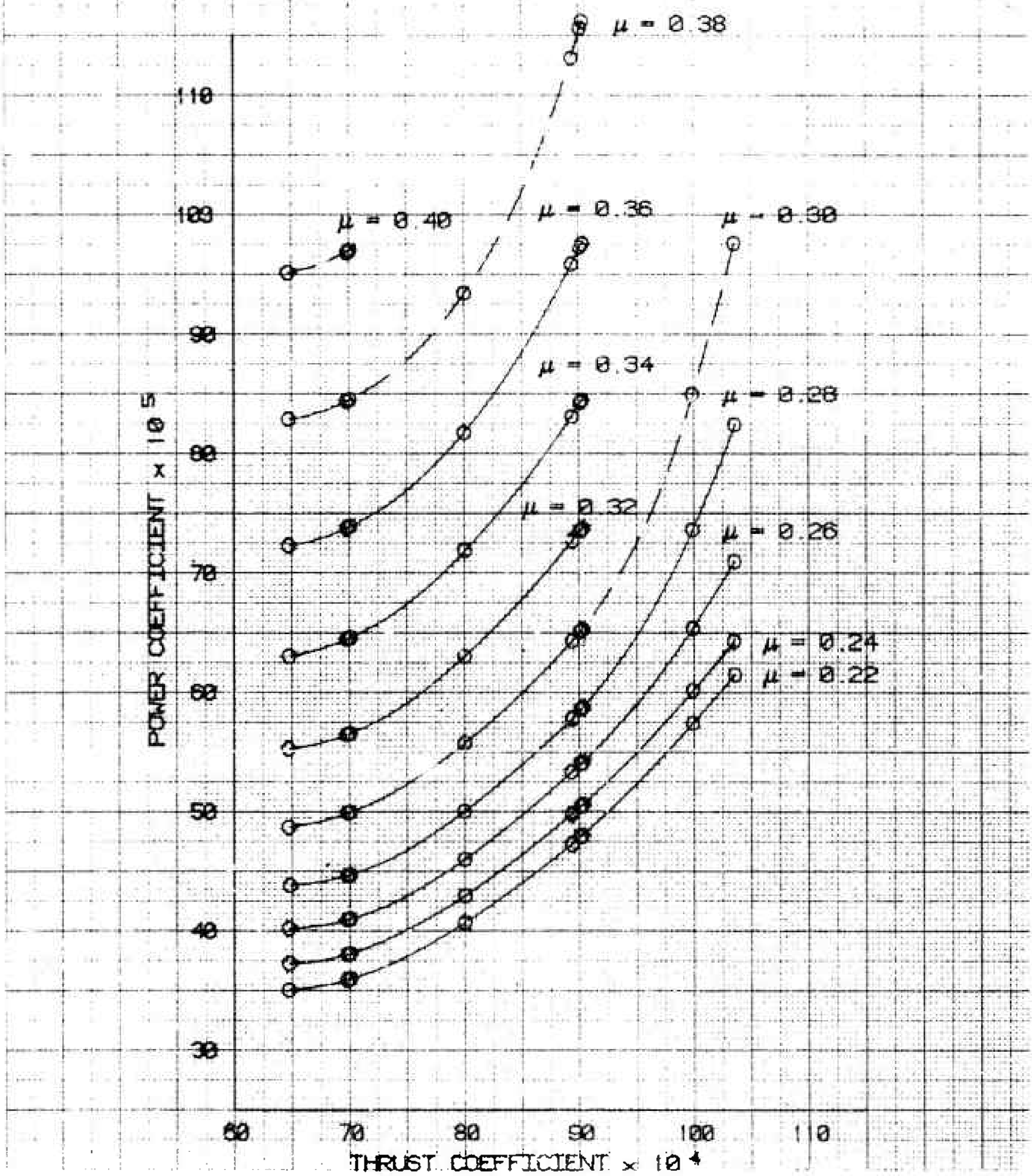


FIGURE 26

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 82-23748

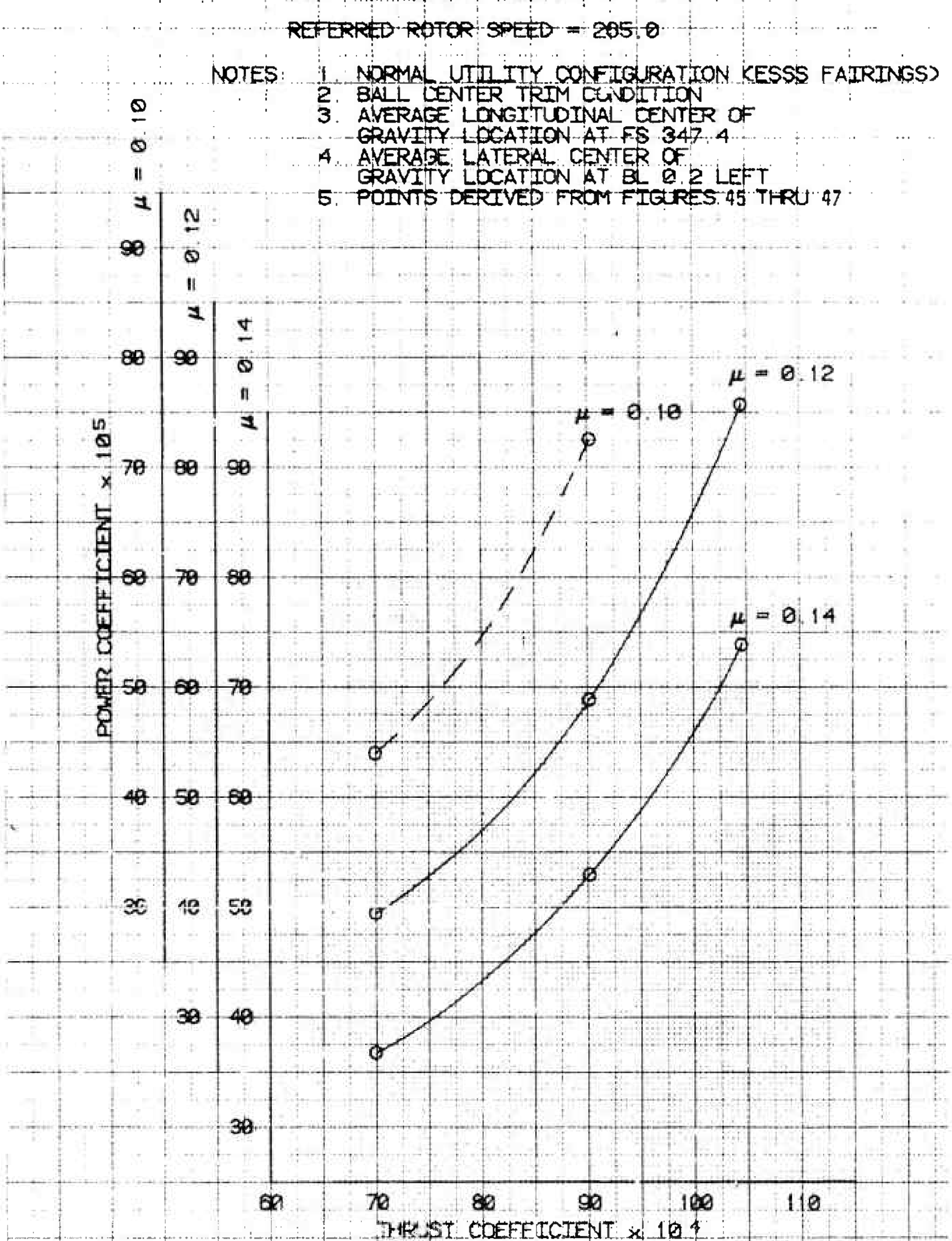


FIGURE 27

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 265.0

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. POINTS DERIVED FROM FIGURES 45 THRU 47

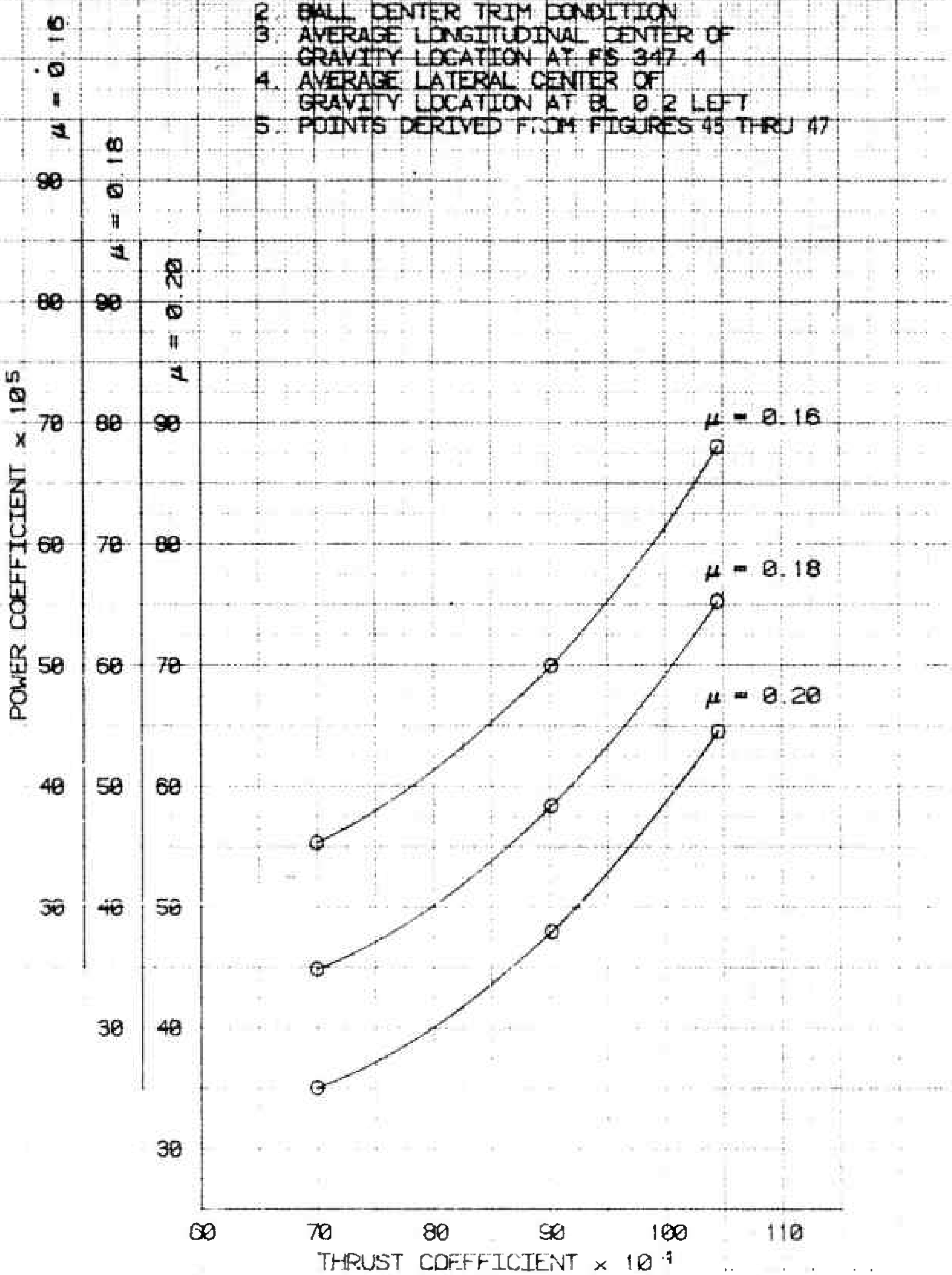


FIGURE 28

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 205.0

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. POINTS DERIVED FROM FIGURES 45 THRU 47

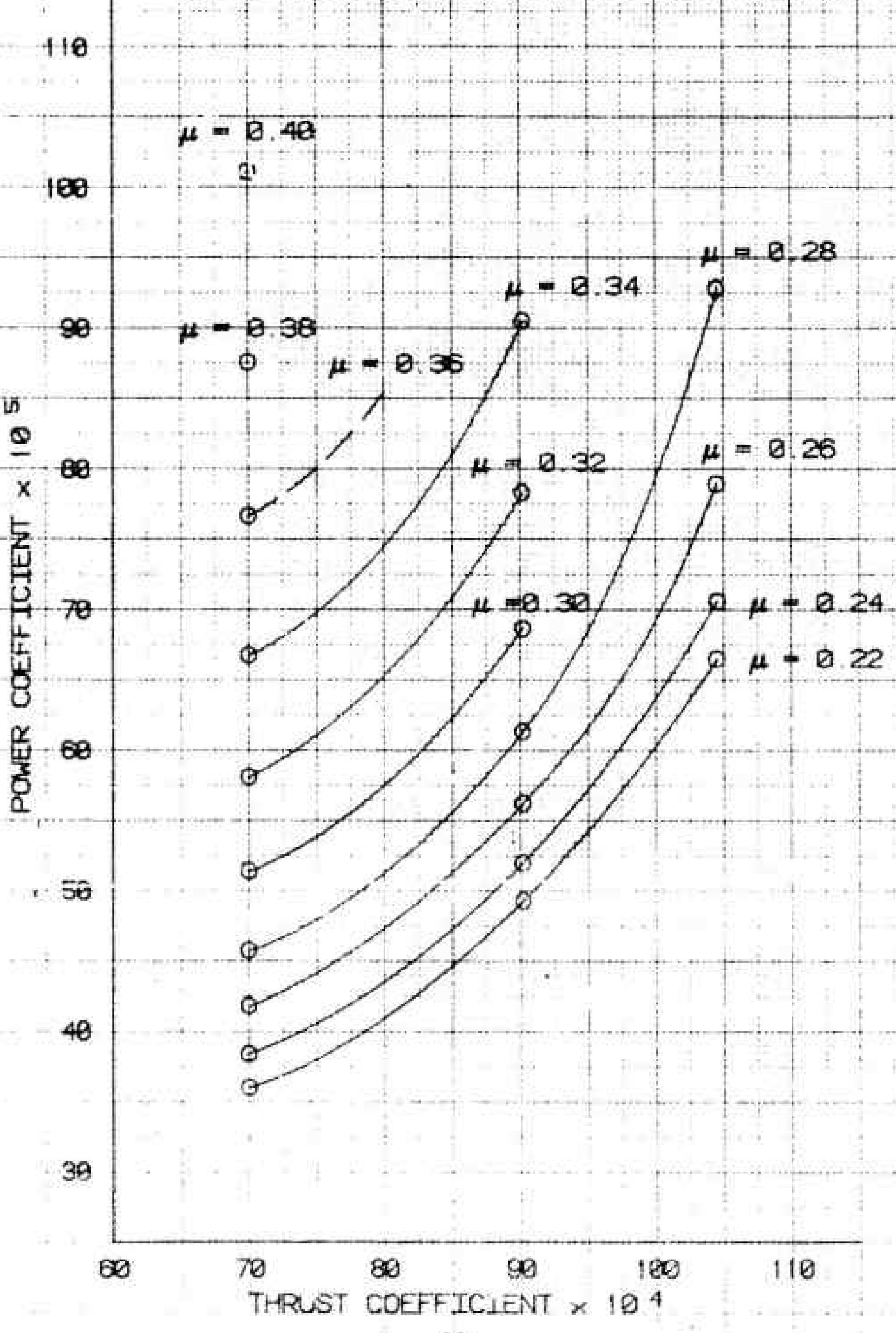


FIGURE 29
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 275.1

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.1 LEFT
5. POINTS DERIVED FROM FIGURES 48 THRU 52

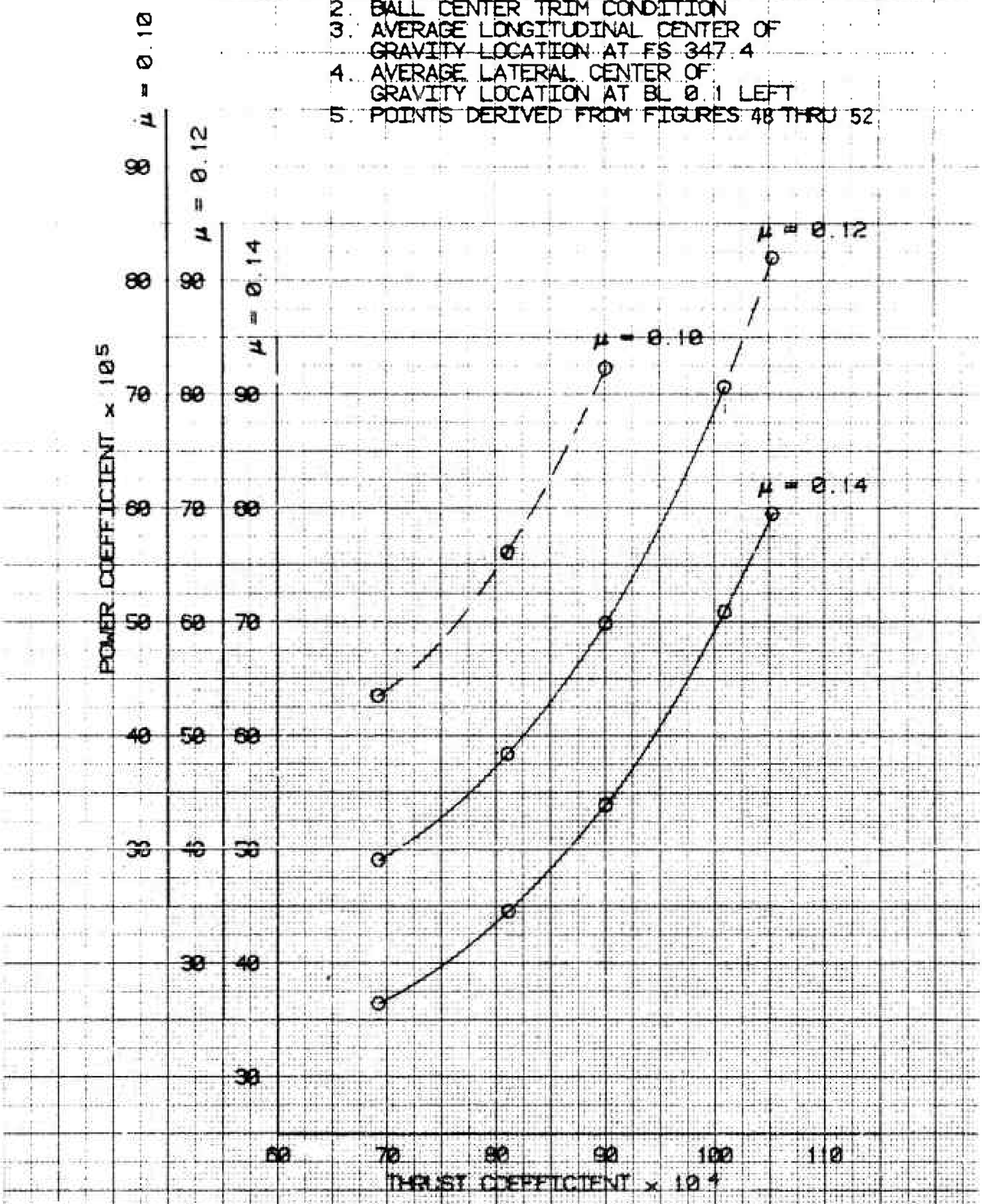


FIGURE 30
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 275.1

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.1 LEFT
5. POINTS DERIVED FROM FIGURES 48 THRU 52

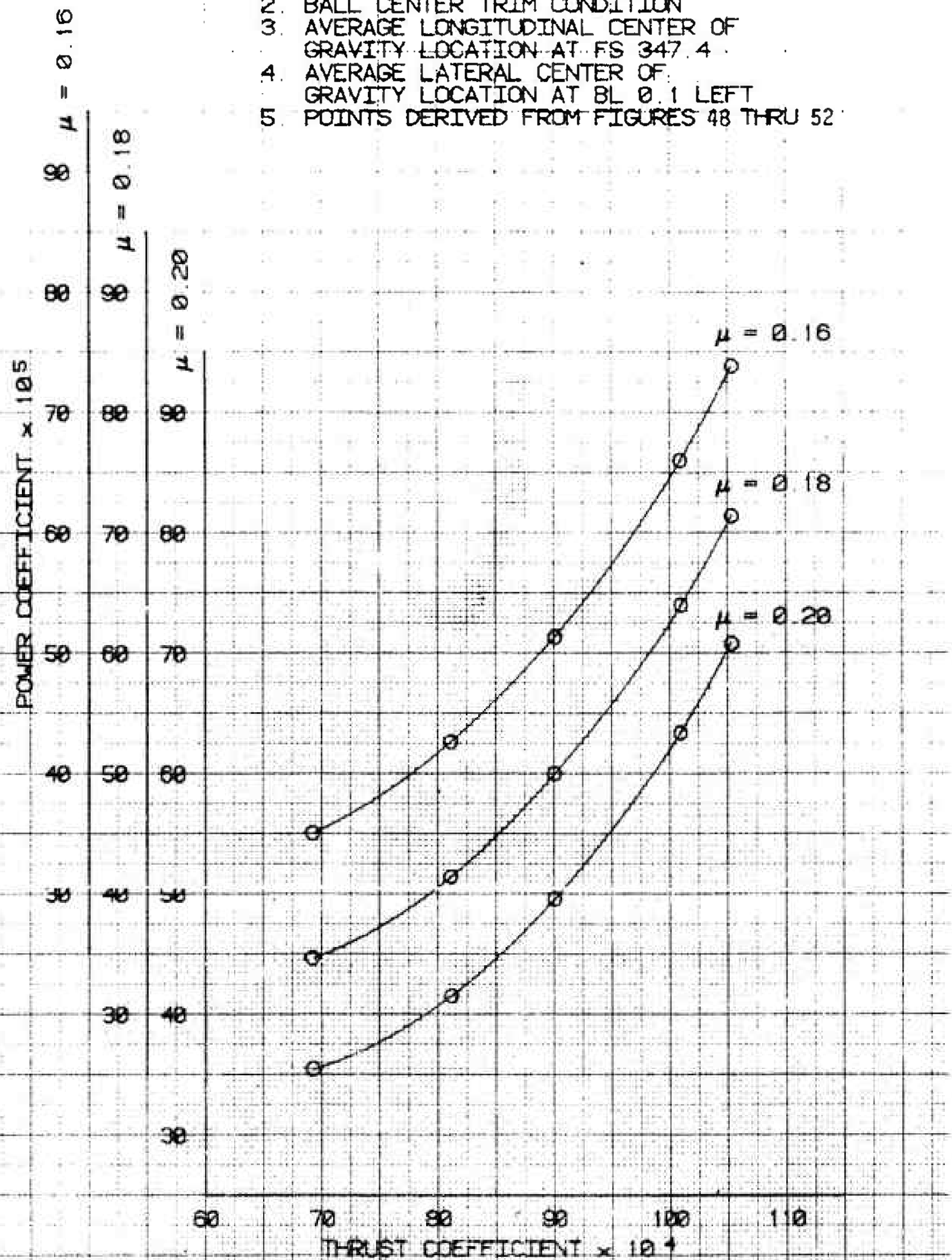


FIGURE 31
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

REFERRED ROTOR SPEED = 275.1

NOTES:
1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTER TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.4
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.1 LEFT
5. POINTS DERIVED FROM FIGURES 48 THRU 52

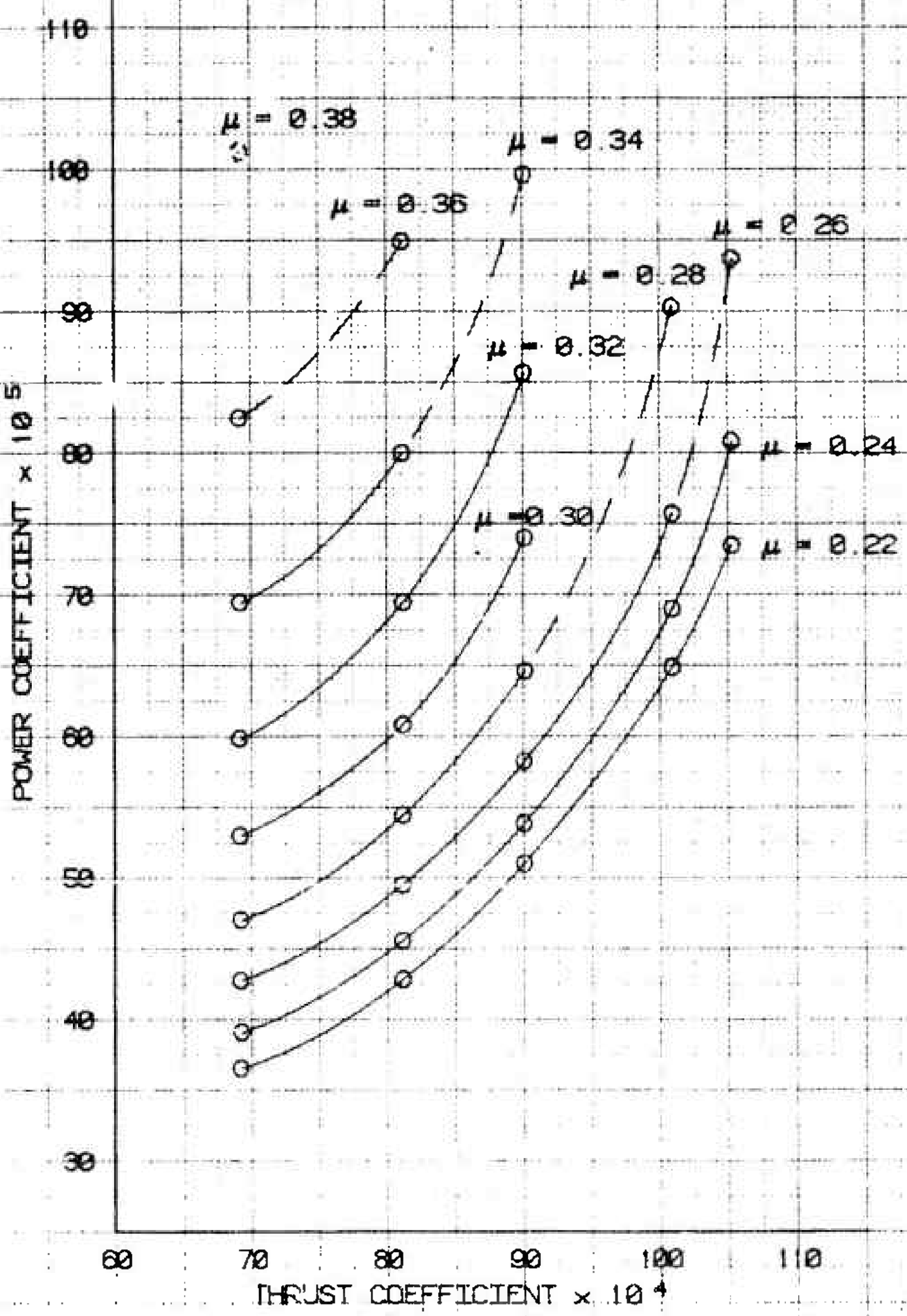


FIGURE 32
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15110	346.9	0.4LT	4070	23.5	244.8	0.006510	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

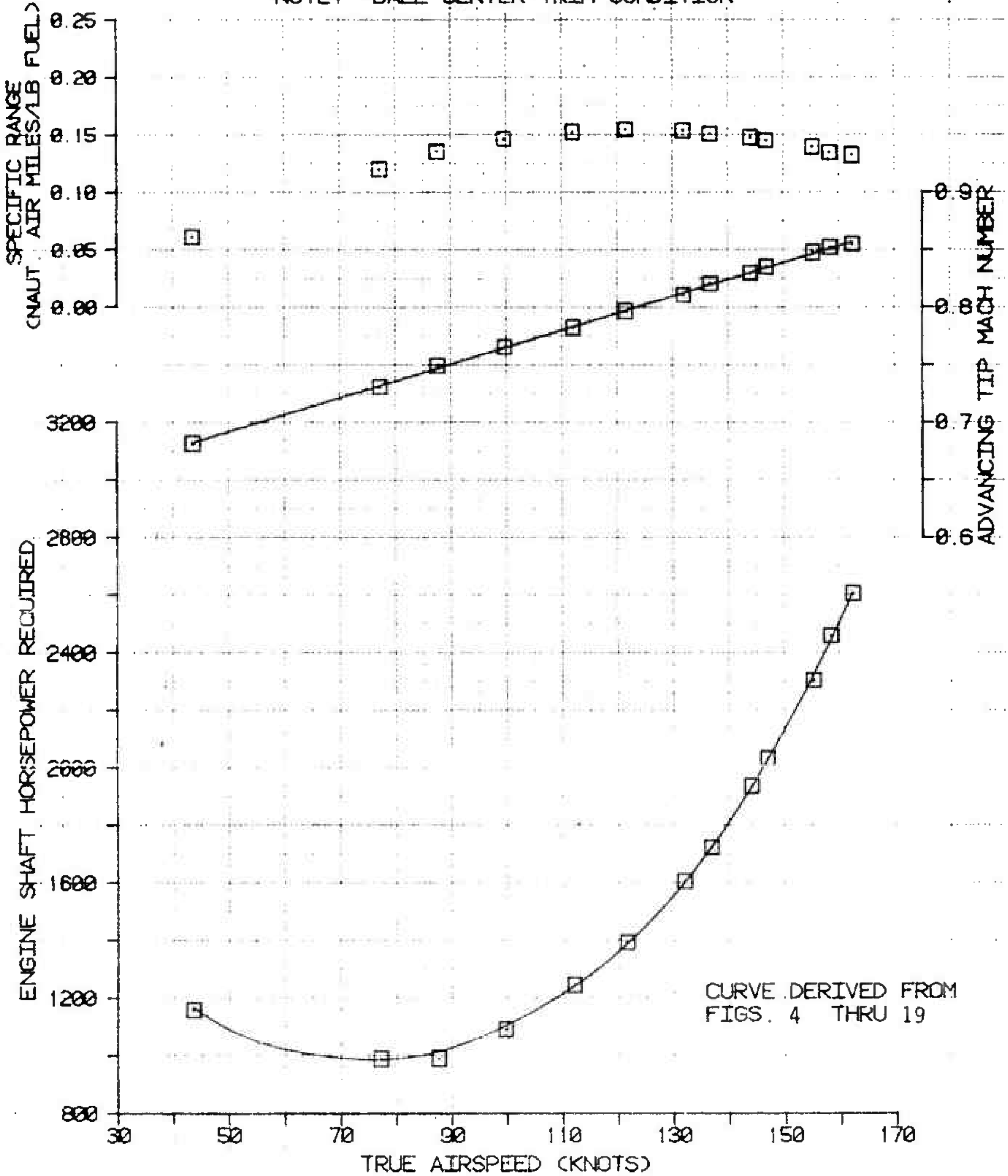


FIGURE 33
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19490	347.1(FWD)	0.2LT	6300	23.0	245.1	0.008978	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

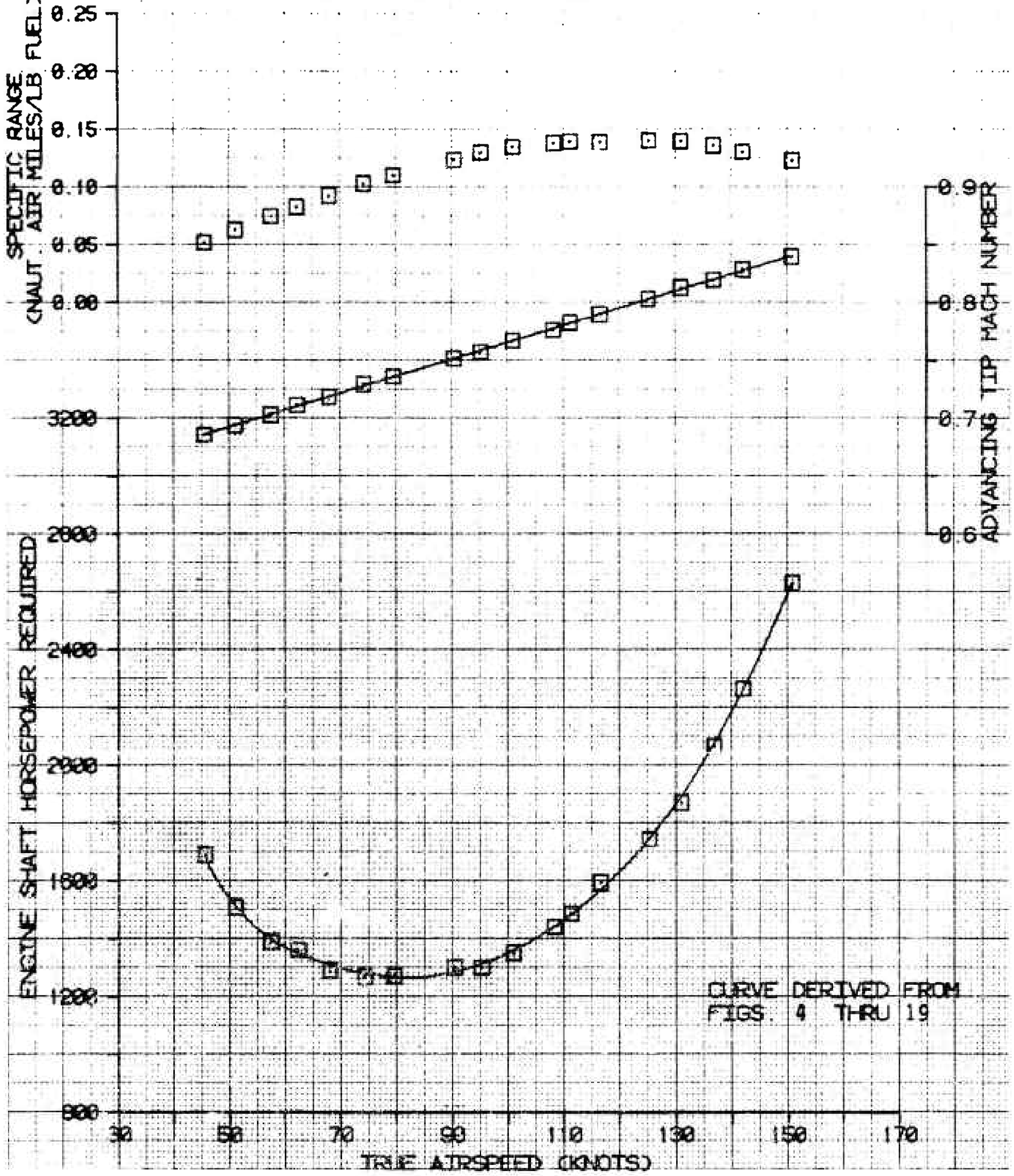
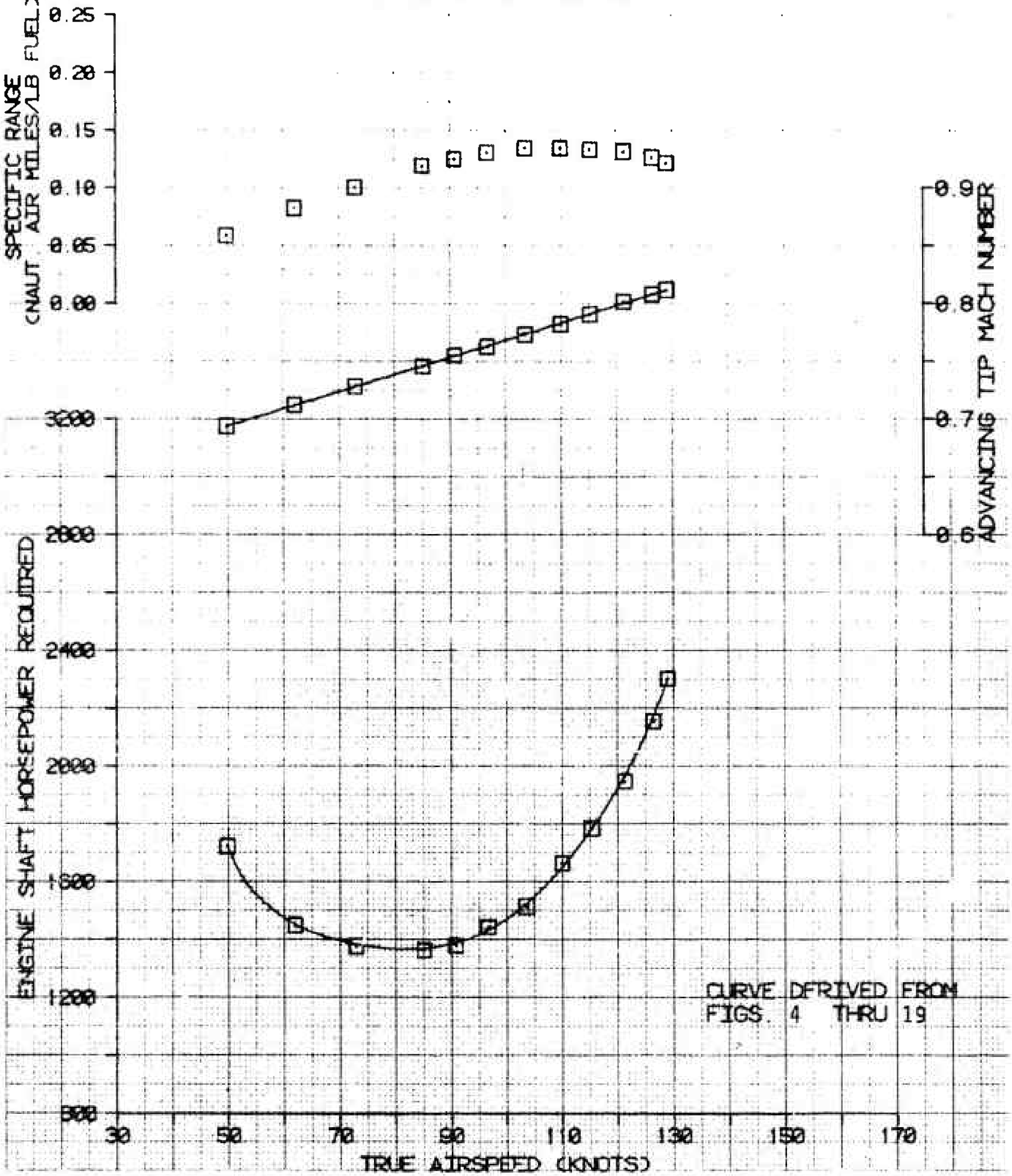


FIGURE 34
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19360	347.3(FWD)	0.3LT	10830	16.5	245.4	0.010472	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION



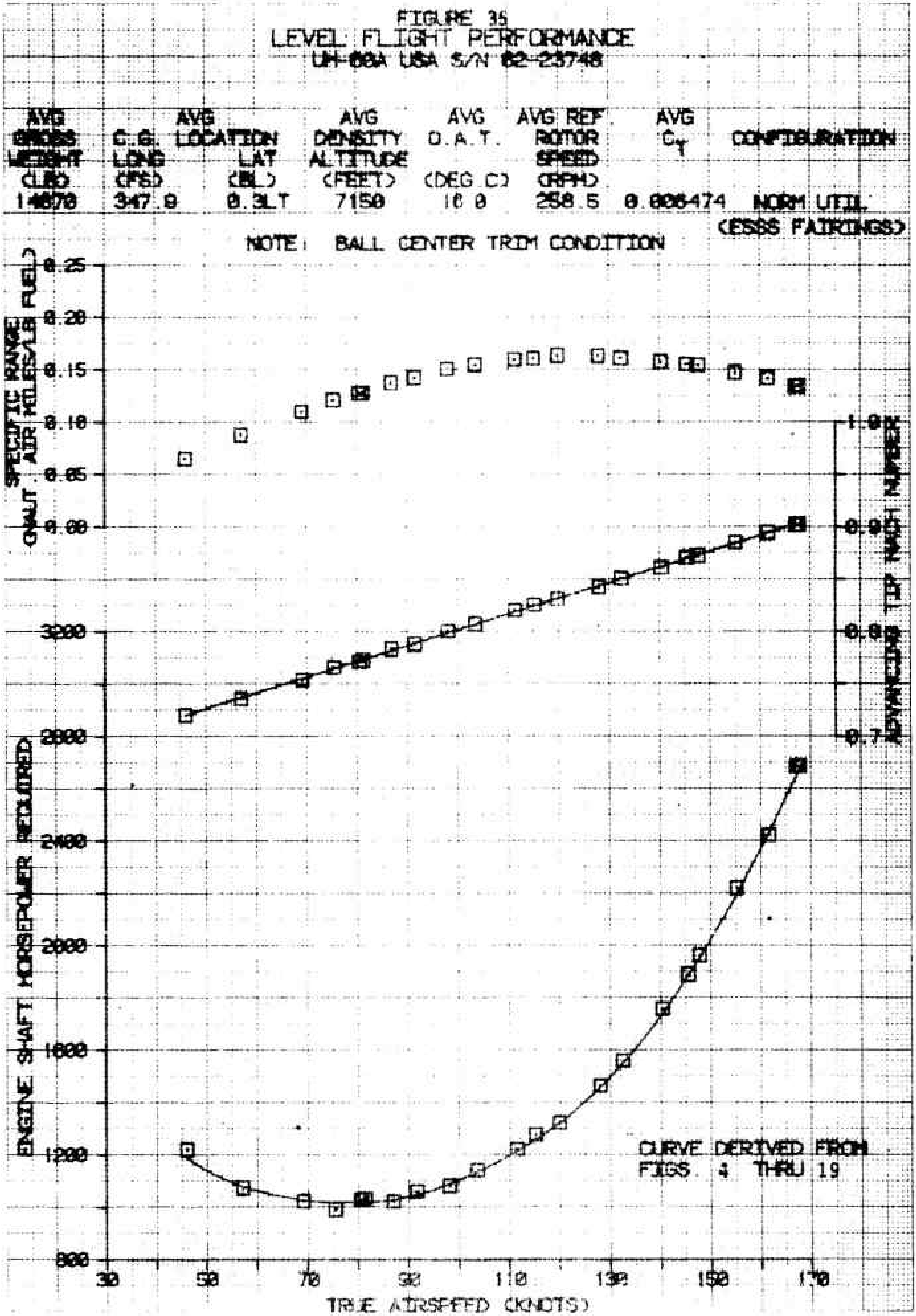


FIGURE 35
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748
AVG GROSS WEIGHT (LB) 14870
AVG C.G. LOCATION LONG (FS) 347.0
LAT (BL) 0.3LT
AVG DENSITY ALTITUDE (FEET) 7150
AVG O.A.T. (DEG C) 10.0
AVG REF. ROTOR SPEED (RPM) 258.5
AVG C_T 0.006474
CONFIGURATION NORM UTIL (ESSS FAIRINGS)
NOTE: BALL CENTER TRIM CONDITION
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
ADVANCING TIP MACH NUMBER
ENGINE SHAFT HORSEPOWER REQUIRED
CURVE DERIVED FROM FIGS. 4 THRU 19
TRUE AIRSPEED (KNOTS)

FIGURE 36
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15950	348.1	0.0	6880	9.5	258.3	0.006901	NORM UTIL (NOTE 2.)

NOTES: 1. BALL CENTER TRIM CONDITION
2. AN/ALQ-144(V) AND M-130 BRACKETS ADDED
3. 3.5 FT^2 ΔFE INCORPORATED TO ASSIMILATE DRAG DIFFERENCES BETWEEN TEST AIRCRAFT

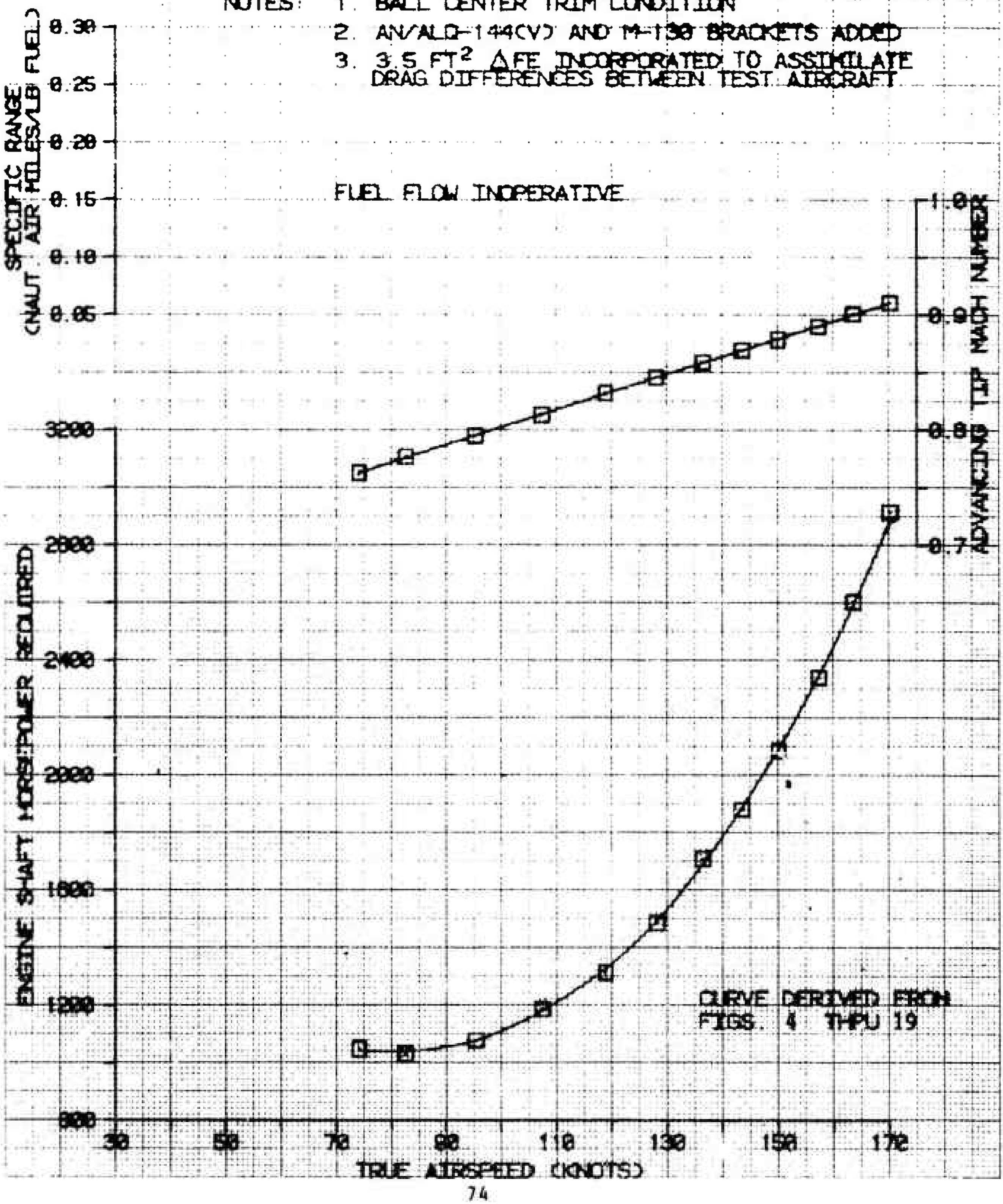


FIGURE 37
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
14470	347.0	0.1LT	10290	14.5	258.0	0.007010	NORM UTIL

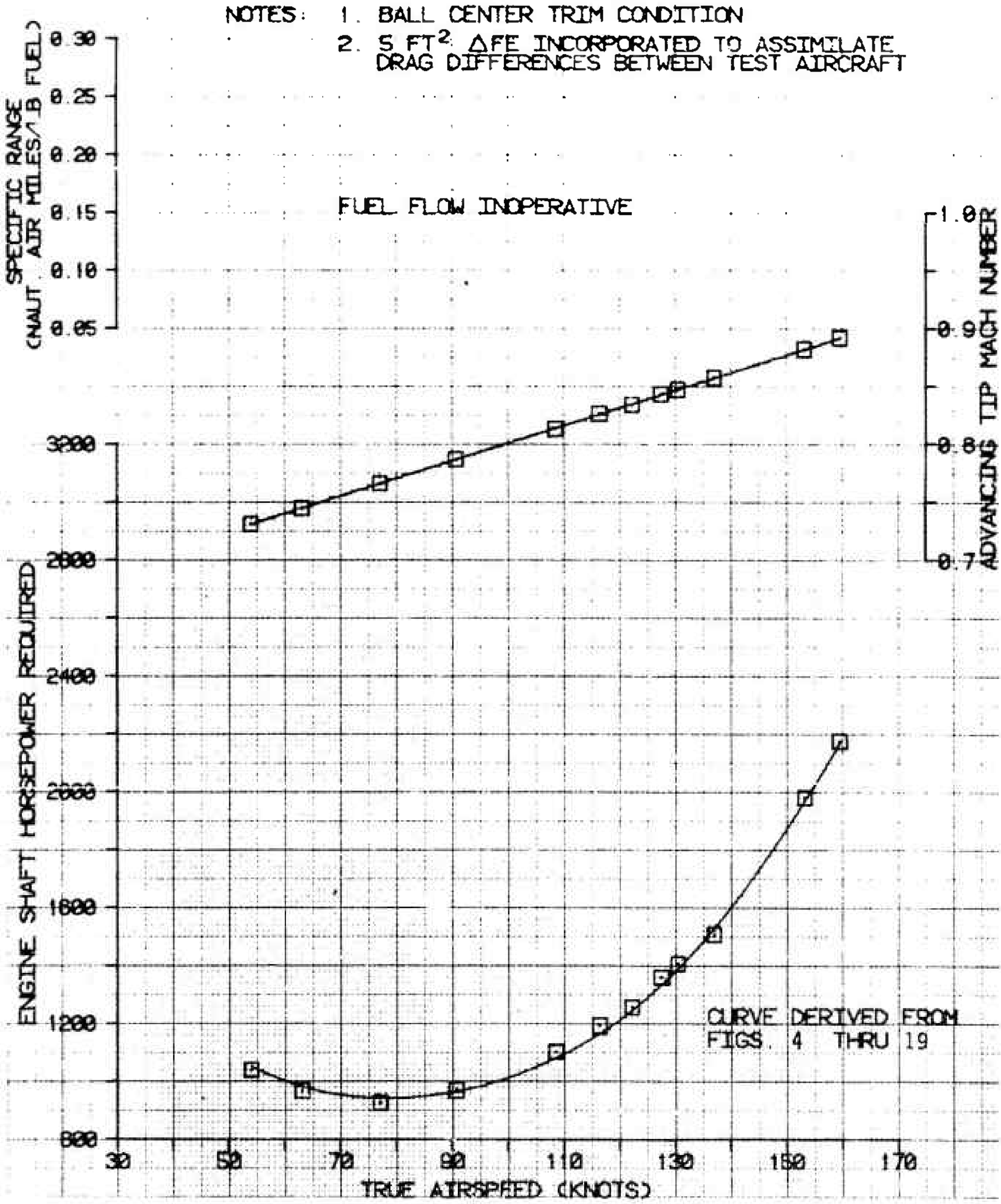


FIGURE 38
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19720	347.5(FWD)	0.3LT	4790	17.0	257.8	0.008005	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

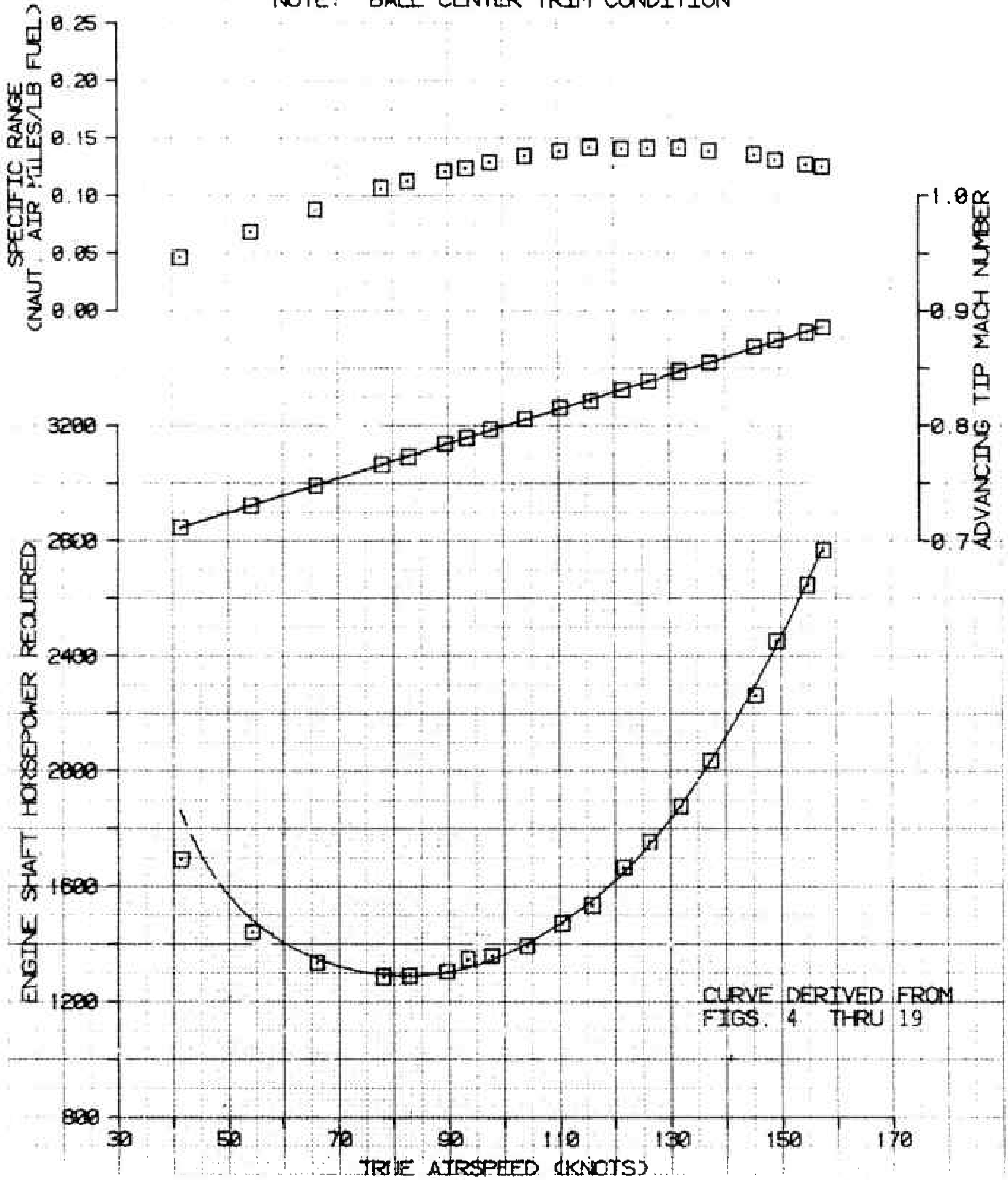


FIGURE 39

LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15920	347.1	0.1LT	11520	14.5	258.2	0.008006	NORM UTIL

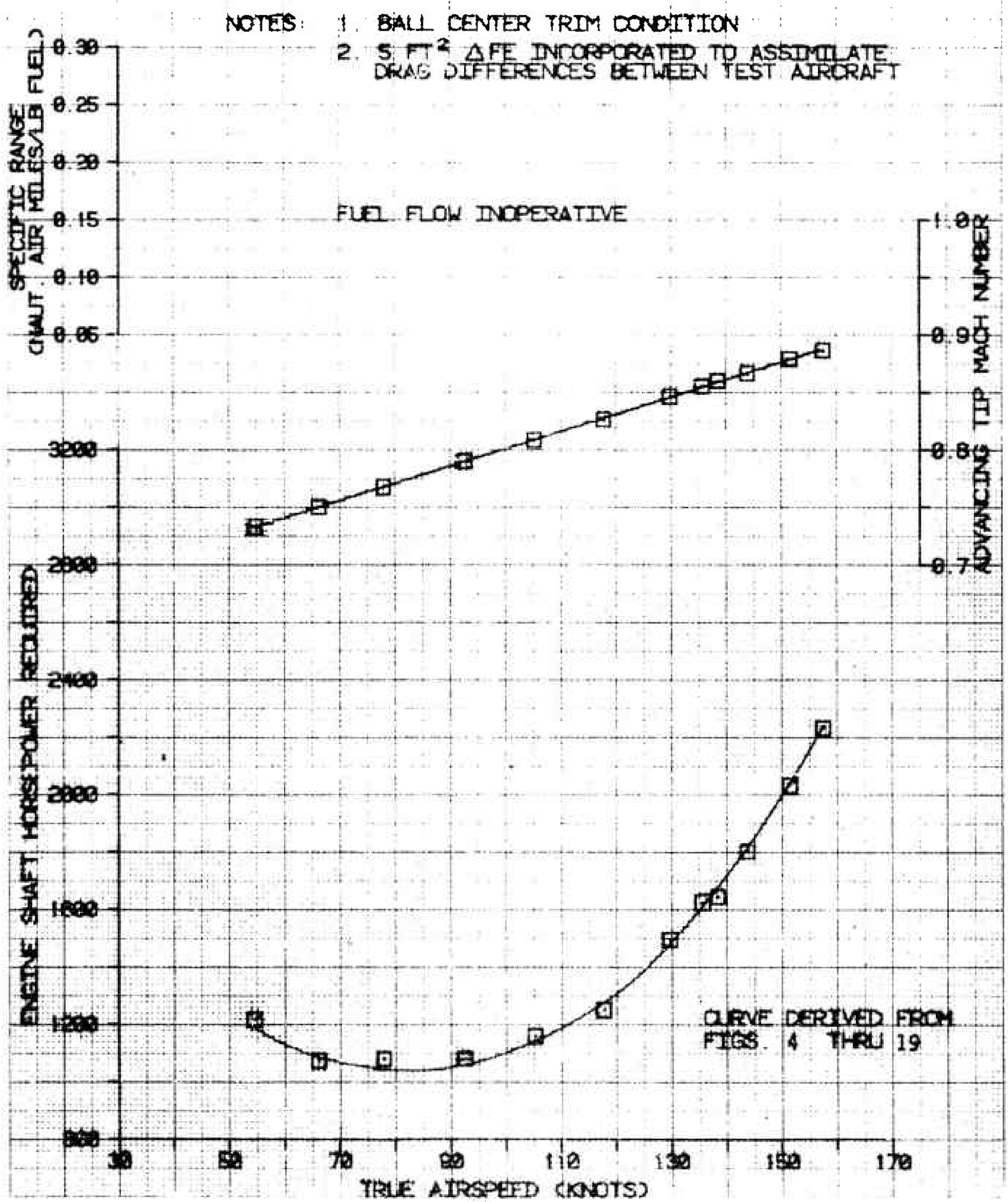


FIGURE 40
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15840	347.1	0.0	14000	3.0	258.8	0.008943	NORM UTIL (NOTE 2.)

NOTES:
1. BALL CENTER TRIM CONDITION
2. AN/ALQ-144(V) AND M-130 BRACKETS ADDED
3. 3.5 FT^2 ΔFE INCORPORATED TO ASSIMILATE DRAG DIFFERENCES BETWEEN TEST AIRCRAFT

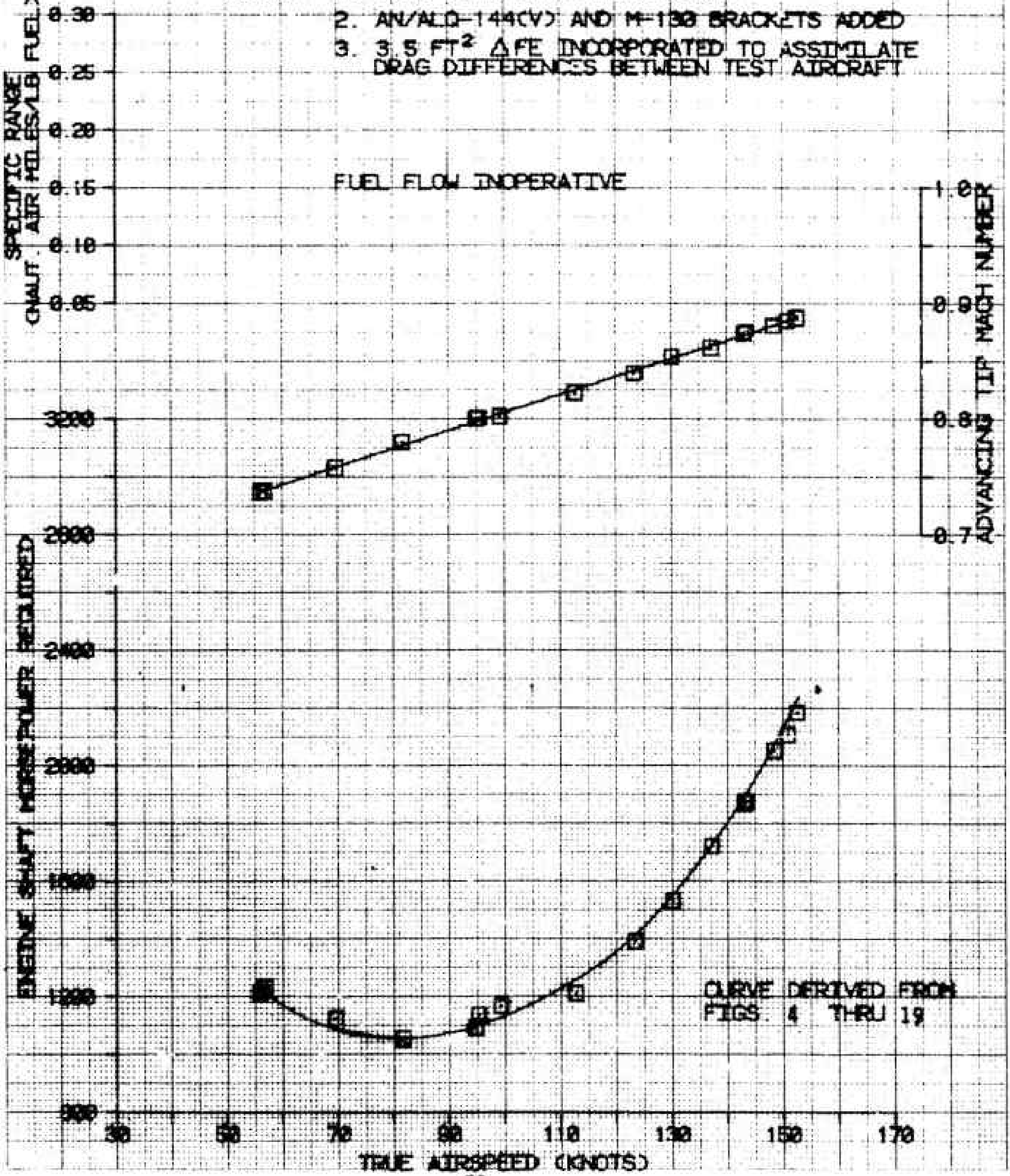


FIGURE 41
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
16190	347.1	0.1LT	13840	7.0	258.0	0.009019	NORM UTIL

NOTES: 1. BALL CENTER TRIM CONDITION
2. 5 FT² ΔFE INCORPORATED TO ASSIMILATE DRAG DIFFERENCES BETWEEN TEST AIRCRAFT

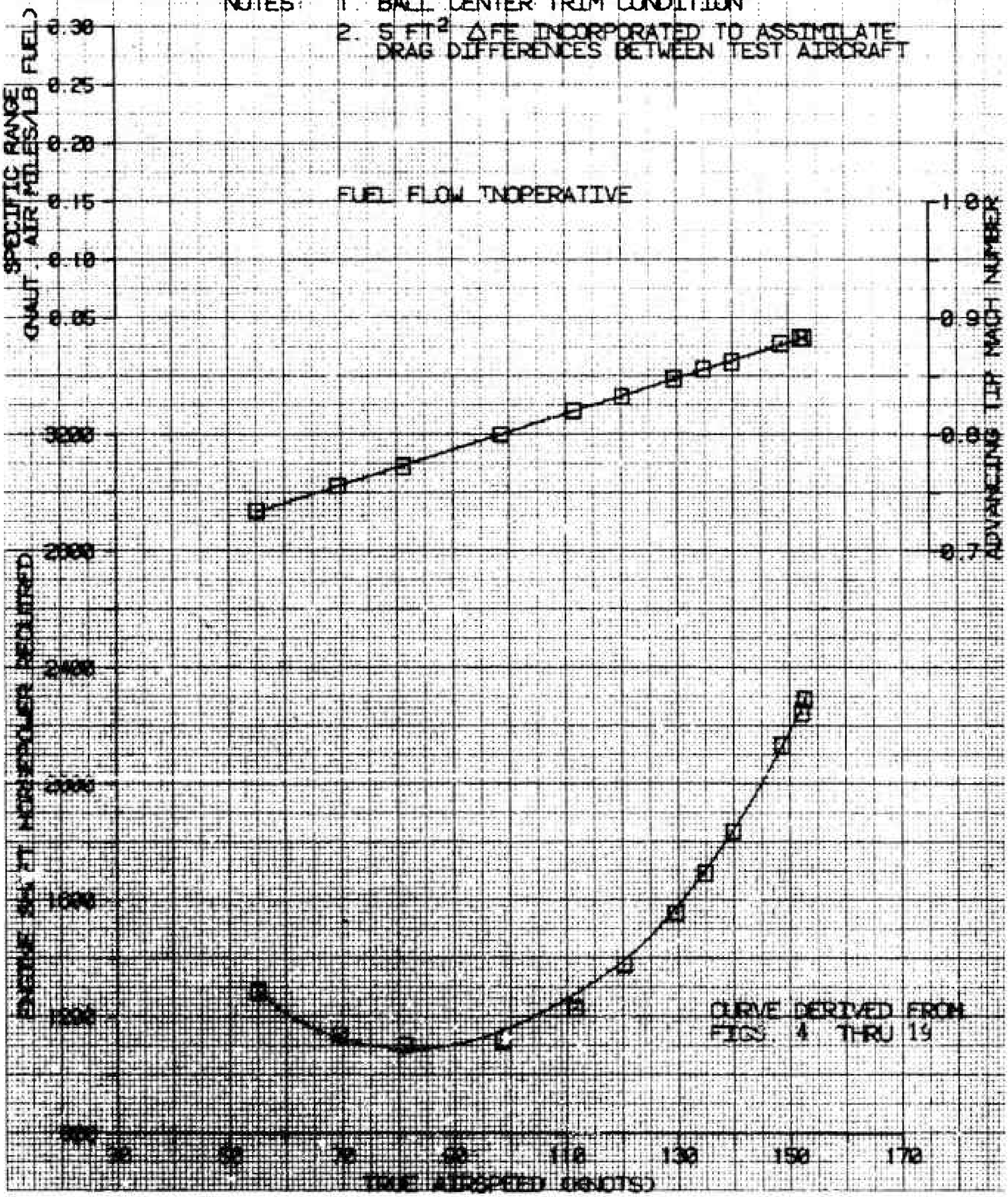


FIGURE 42
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19730	347.8(FWD)	0.3LT	7970	11.0	257.5	0.009035	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

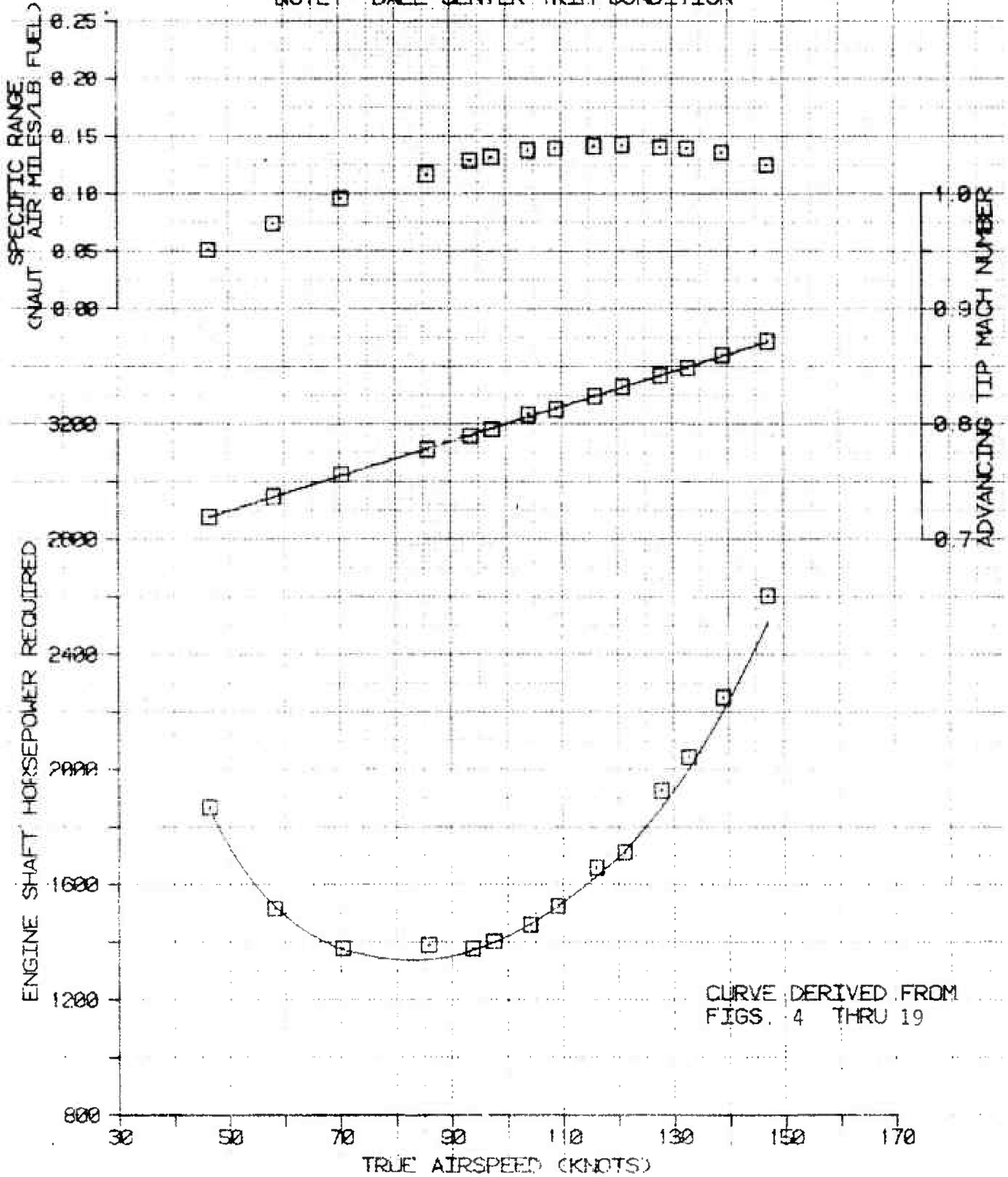


FIGURE 43
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
21690	347.3(FWD)	0.1LT	8870	16.0	258.1	0.009991	NORM UTIL

NOTES: 1. BALL CENTER TRIM CONDITION
2. 5 FT^2 ΔFE INCORPORATED TO ASSIMILATE DRAG DIFFERENCES BETWEEN TEST AIRCRAFT

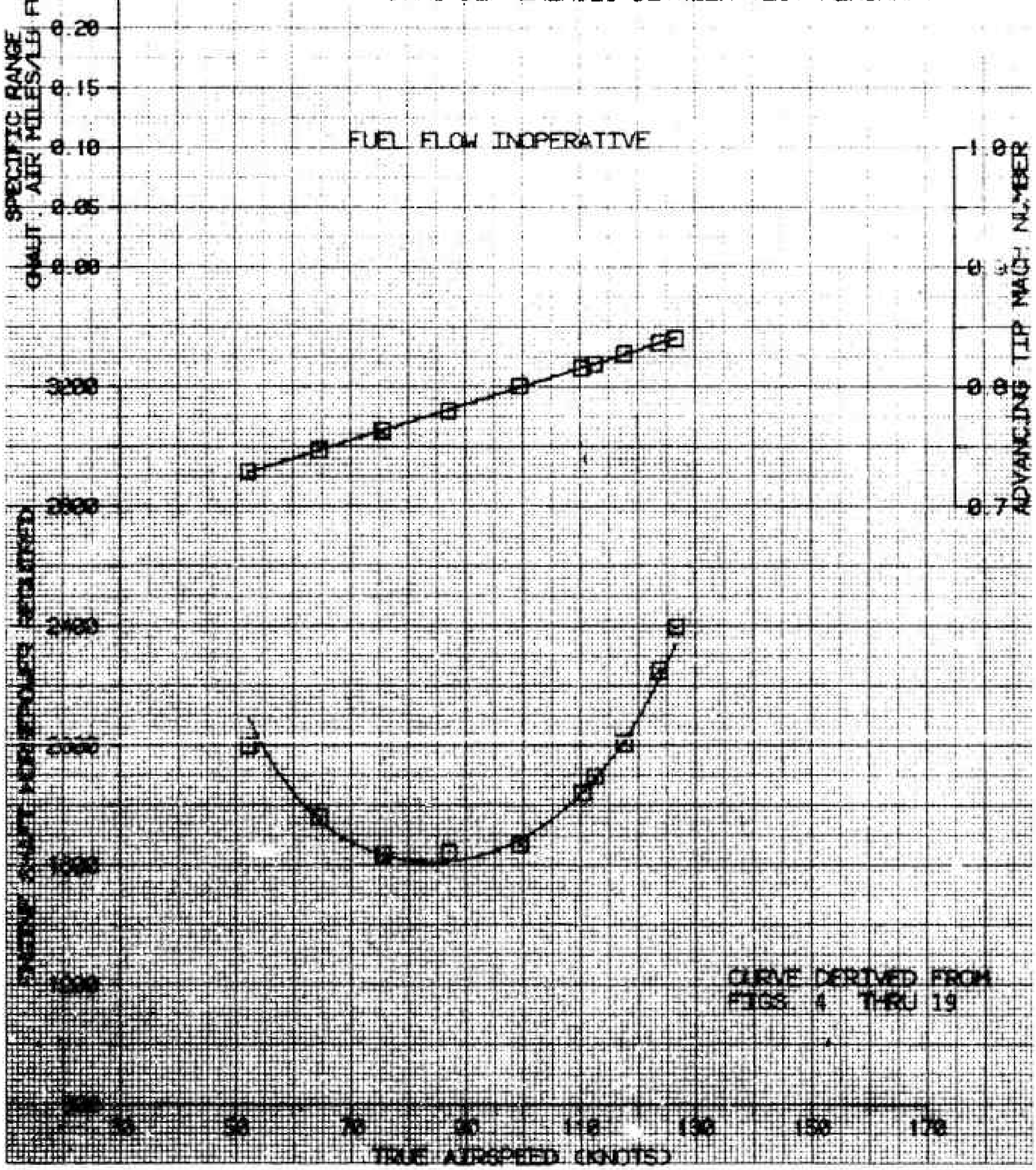


FIGURE 44
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19870	347.2(FWD)	0.3LT	12370	8.5	259.8	0.010357	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

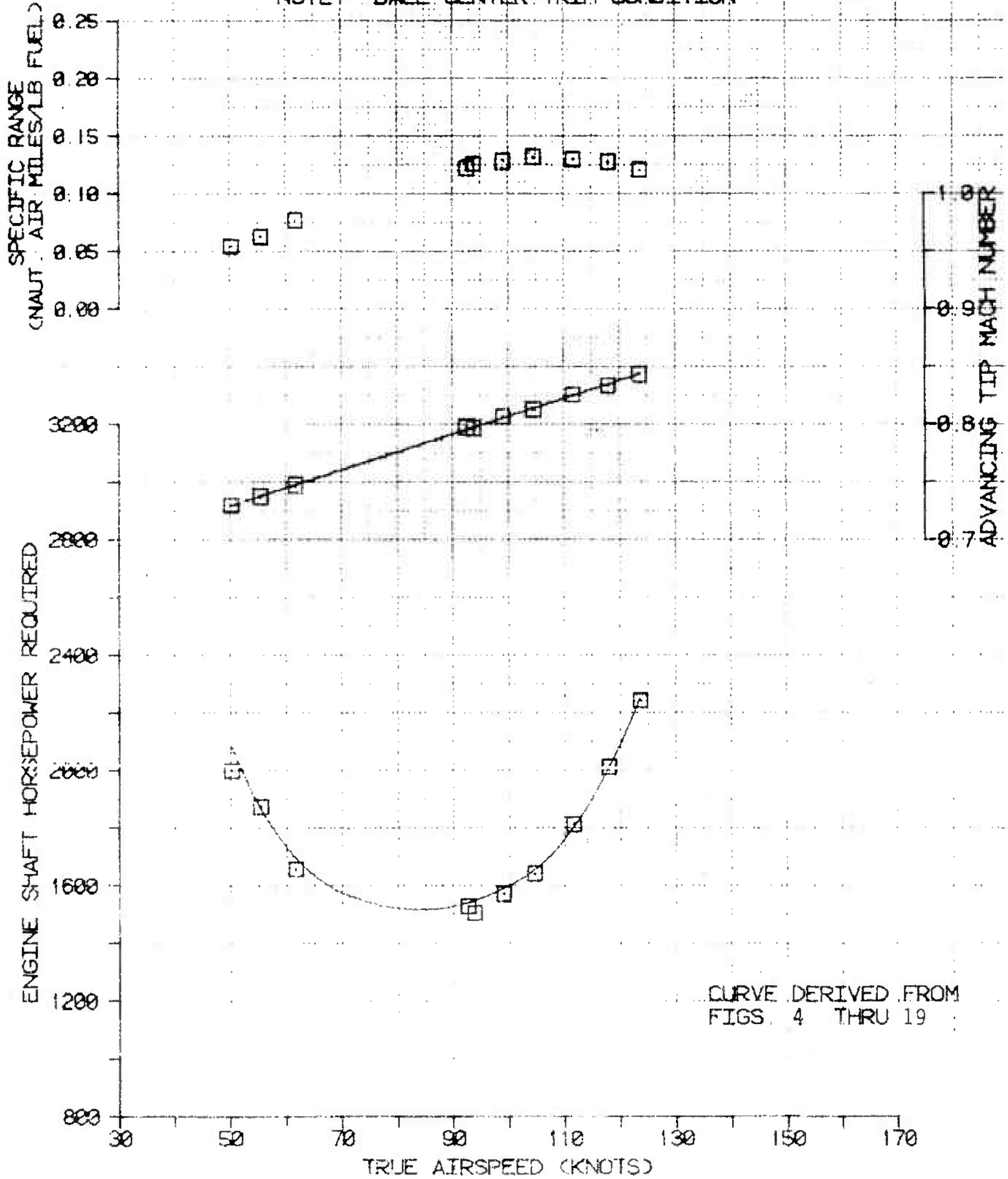


FIGURE 45
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
17000	347.5	0.1LT	3430	-13.0	264.9	0.006997	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

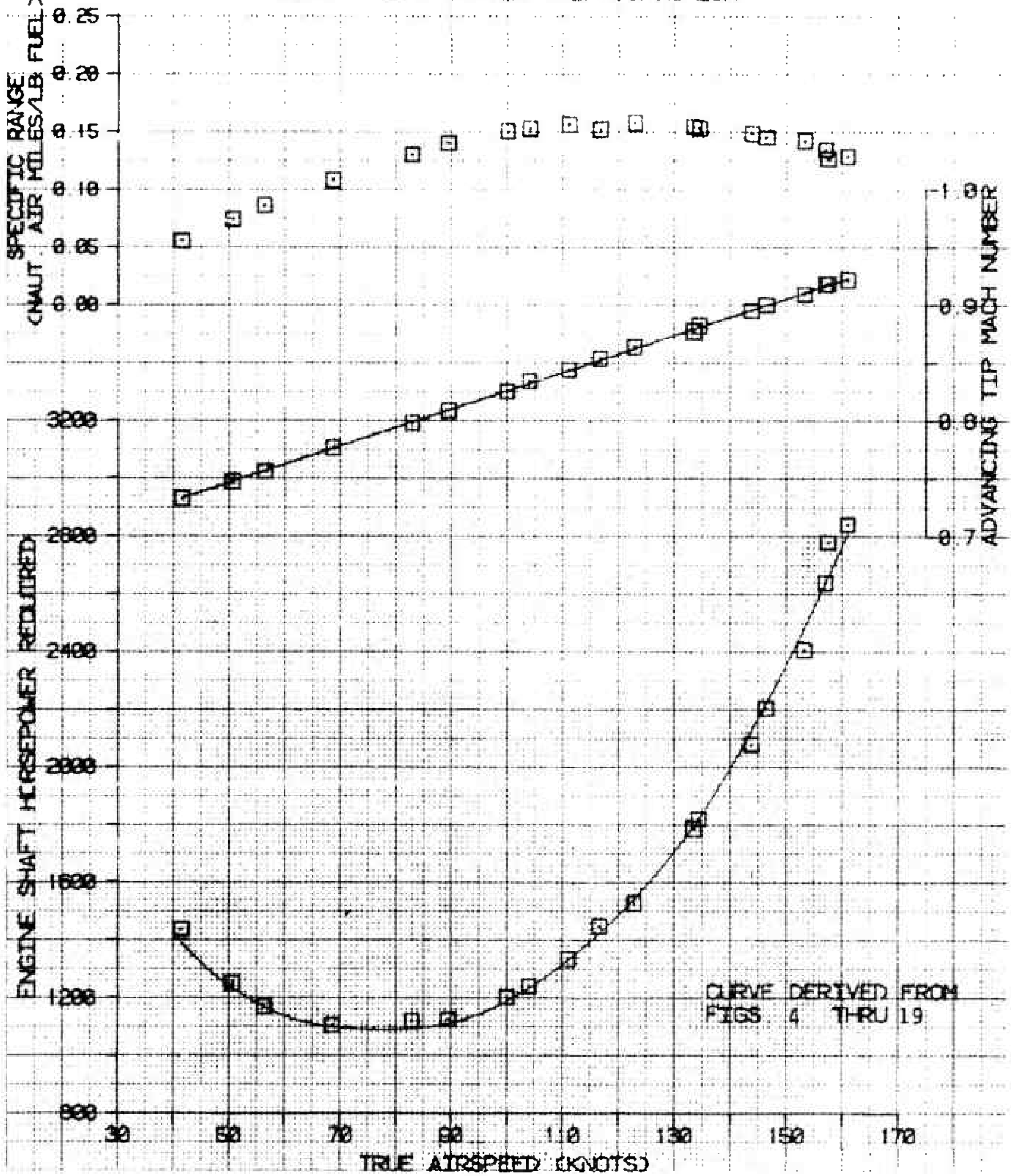


FIGURE 46
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19890	347.2(FWD)	0.3LT	9060	8.0	261.5	0.008022	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

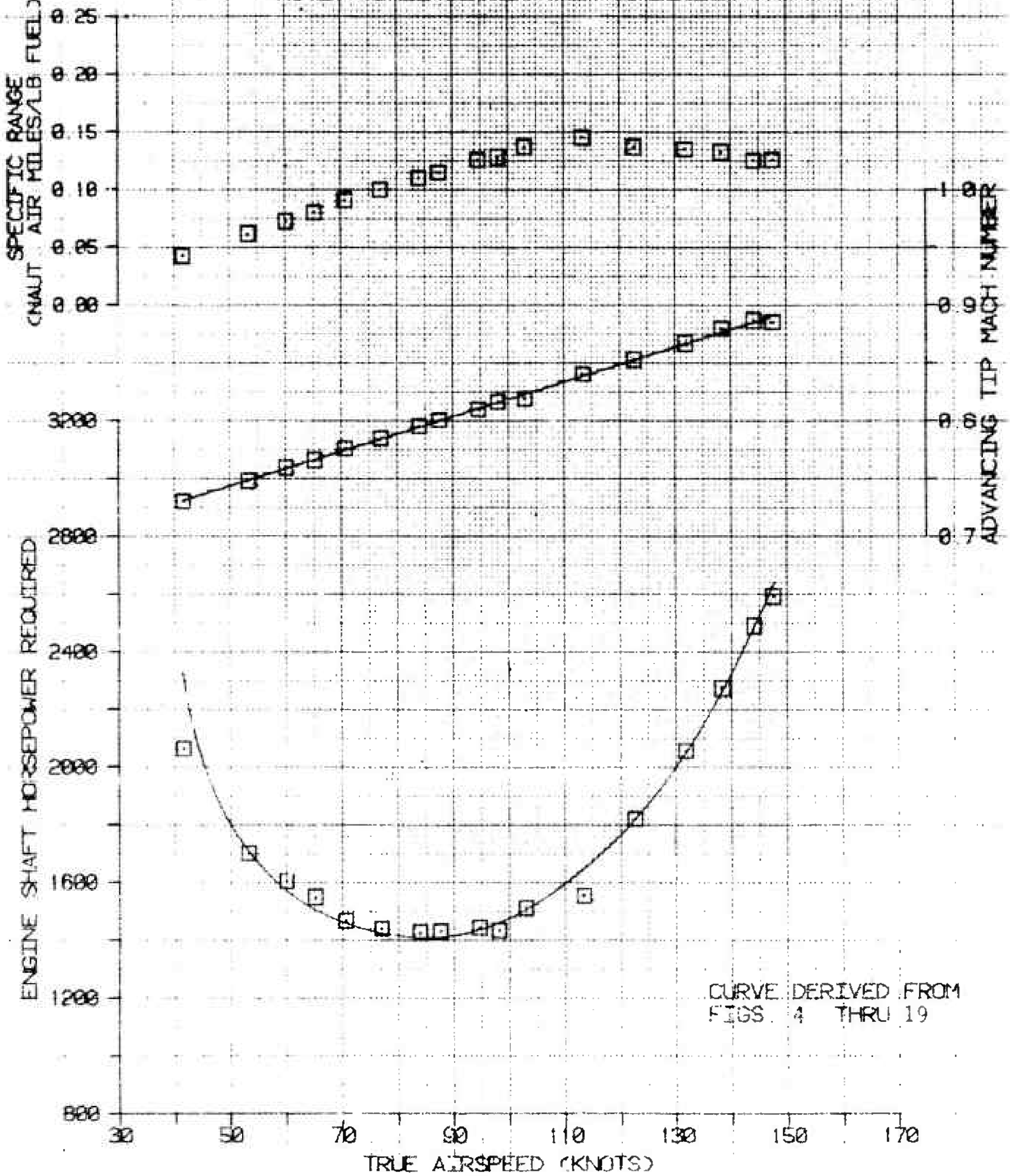


FIGURE 47
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19600	347.5	0.2LT	13880	3.5	265.5	0.010454	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

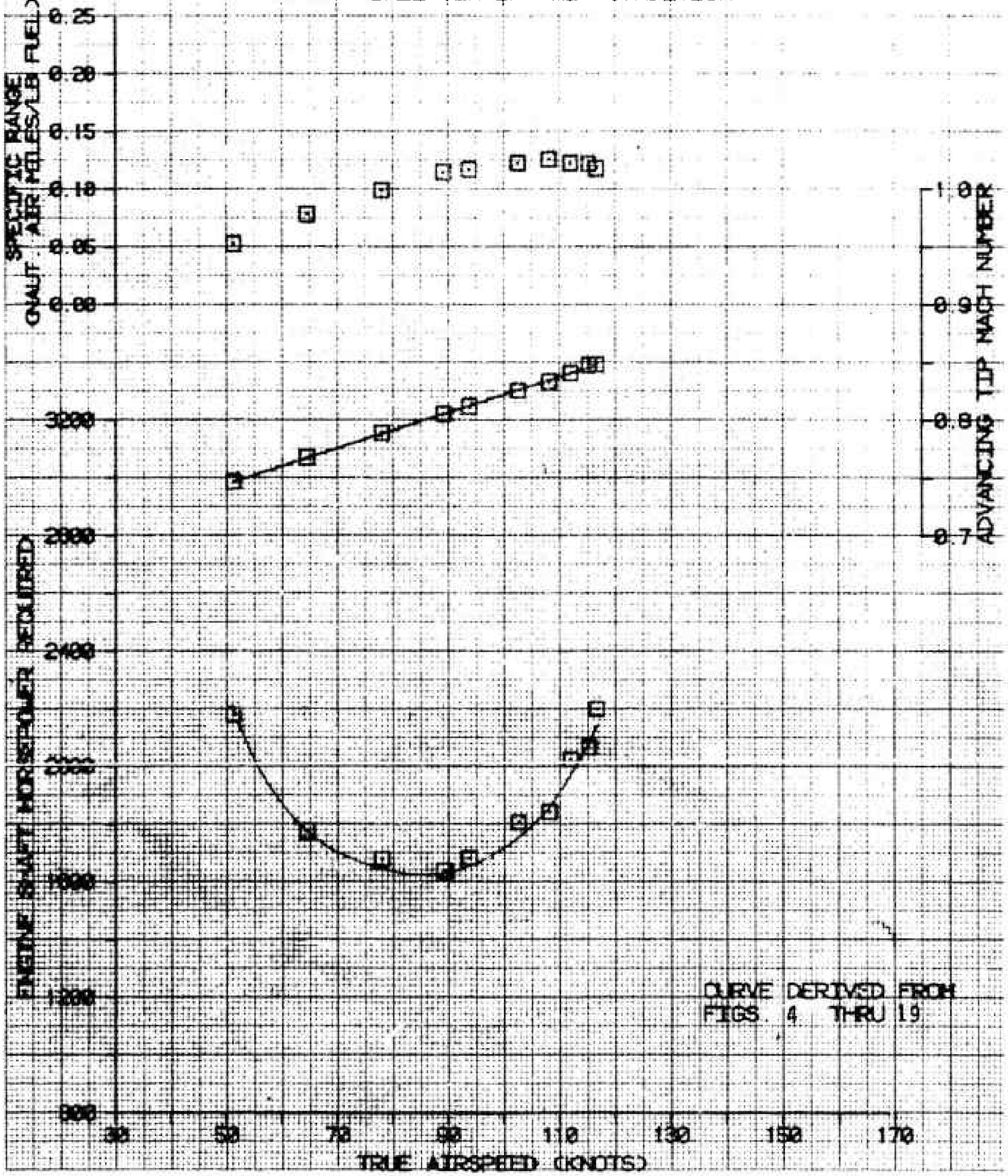


FIGURE 48
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
17080	347.2	0.1LT	5090	-16.0	275.3	0.006922	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION

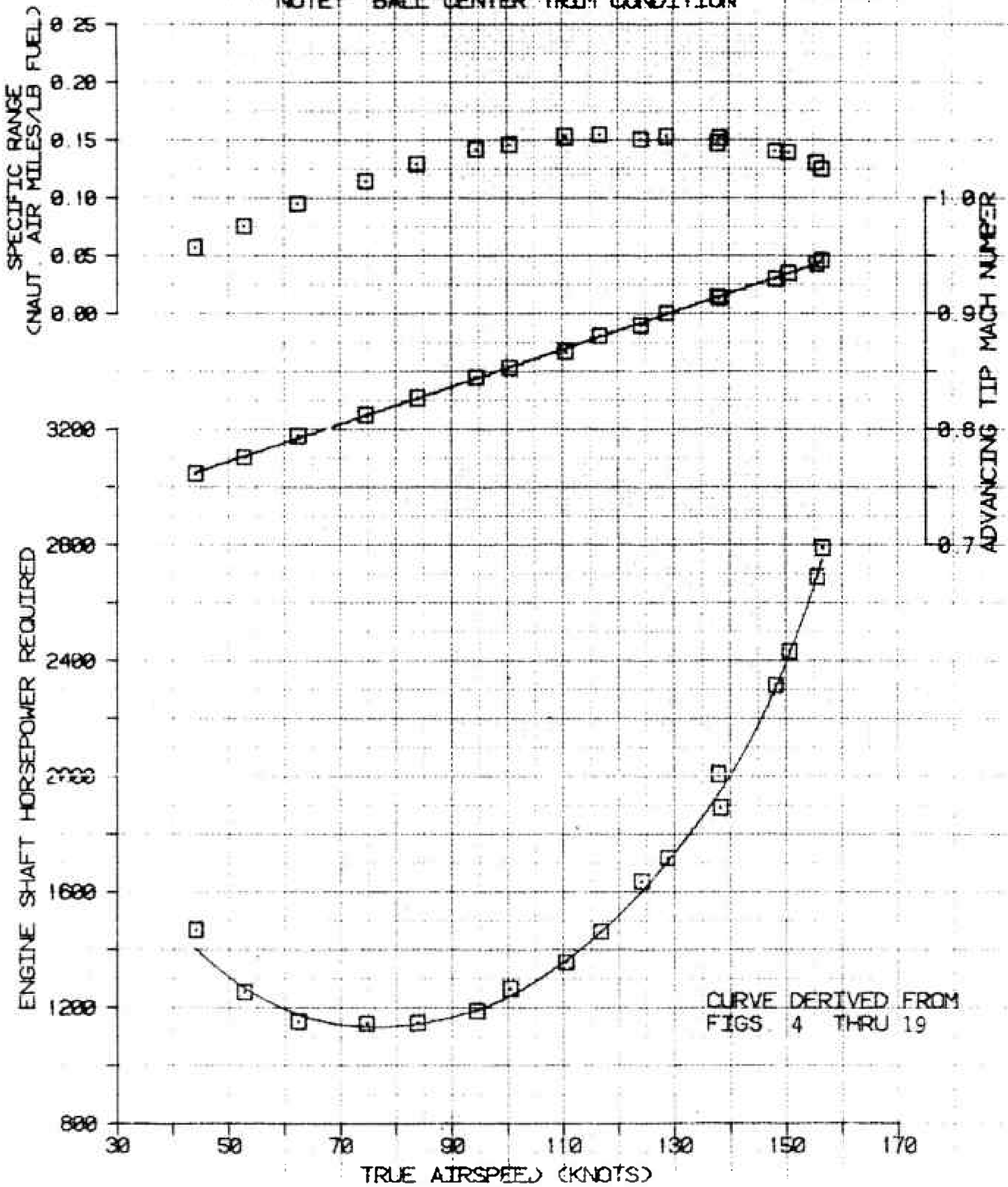


FIGURE 49
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C G LOCATION LONG (FS)	AVG C G LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O A T (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19760	347.7(FWD)	0.1LT	4520	-23.0	274.9	0.008115	NORM UTIL (ESSS FAIRINGS)

NOTE BALL CENTER TRIM CONDITION

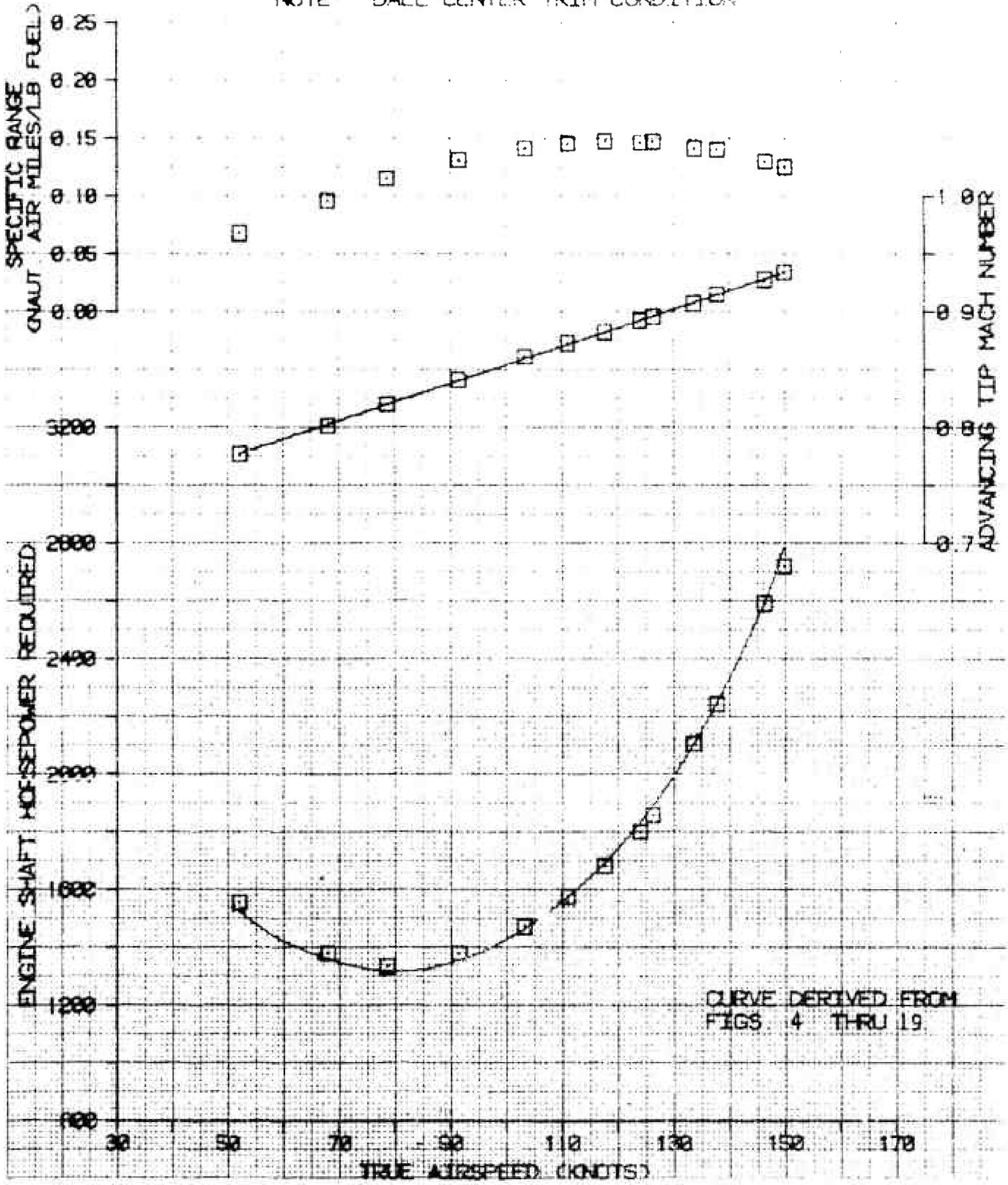


FIGURE 50
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

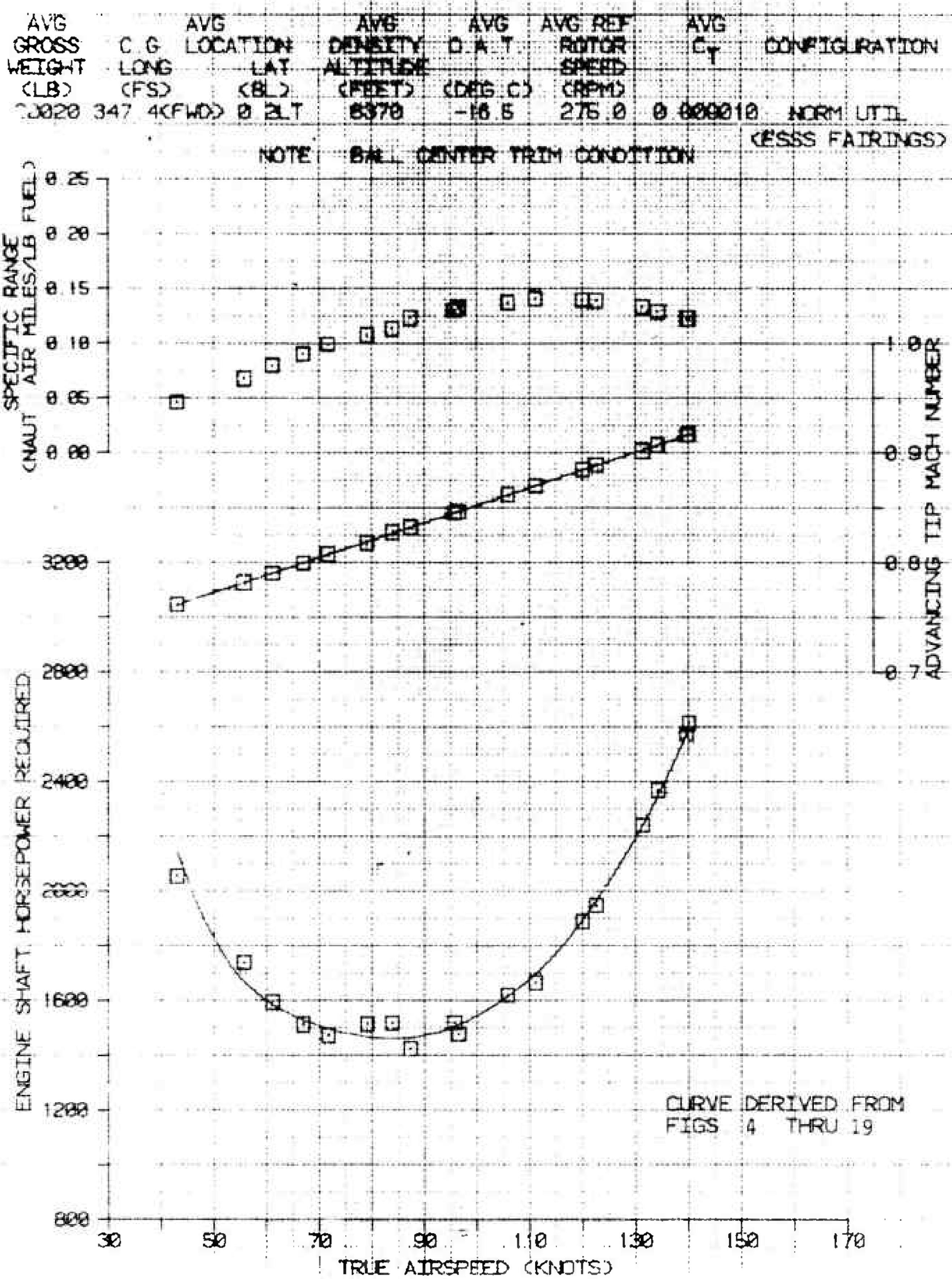
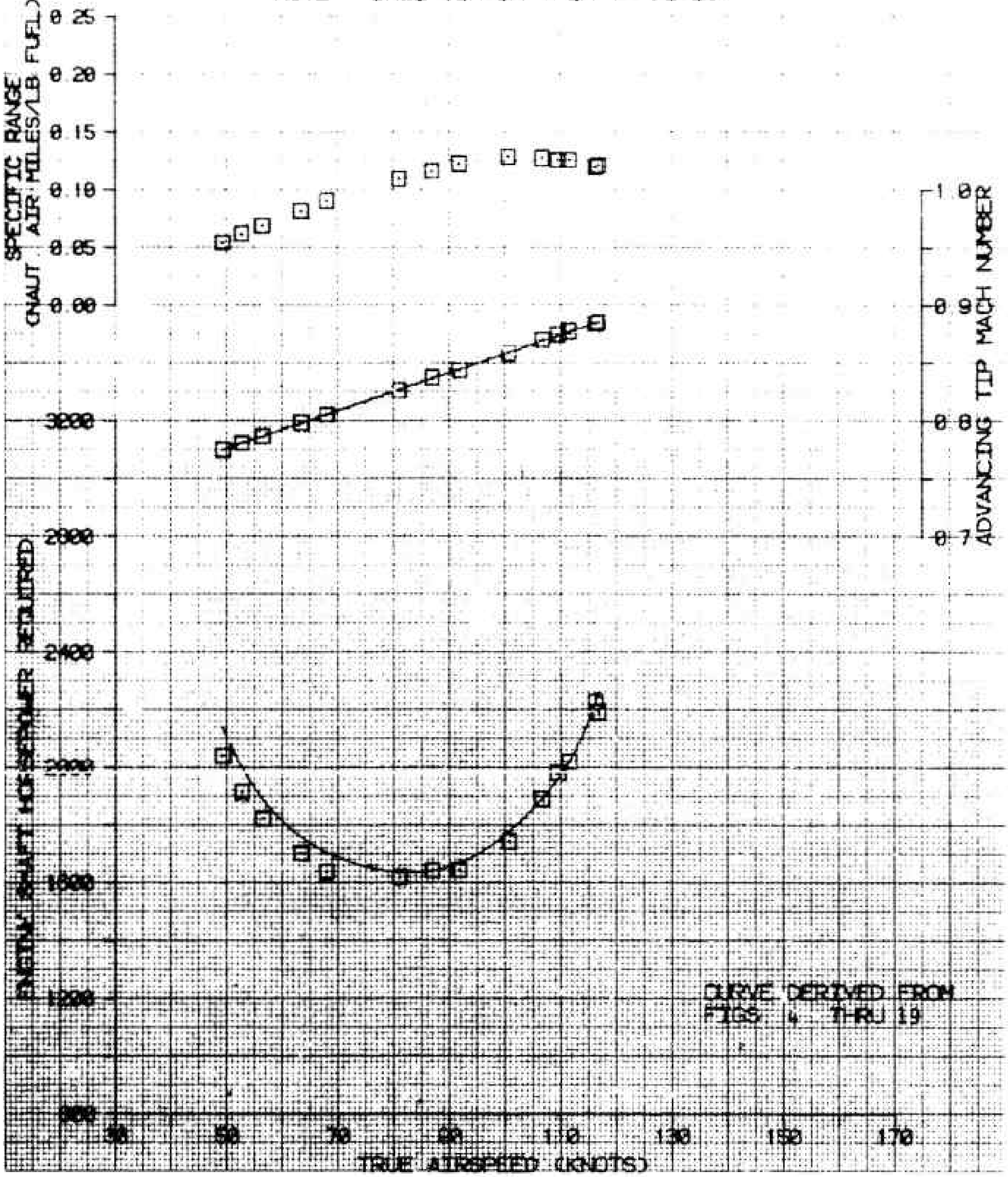


FIGURE 51
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
20160	347.5(FWD)	0.1LT	10200	-30.0	275.7	0.010039	NORM UTIL (ESSS FAIRINGS)

NOTE: BALL CENTER TRIM CONDITION



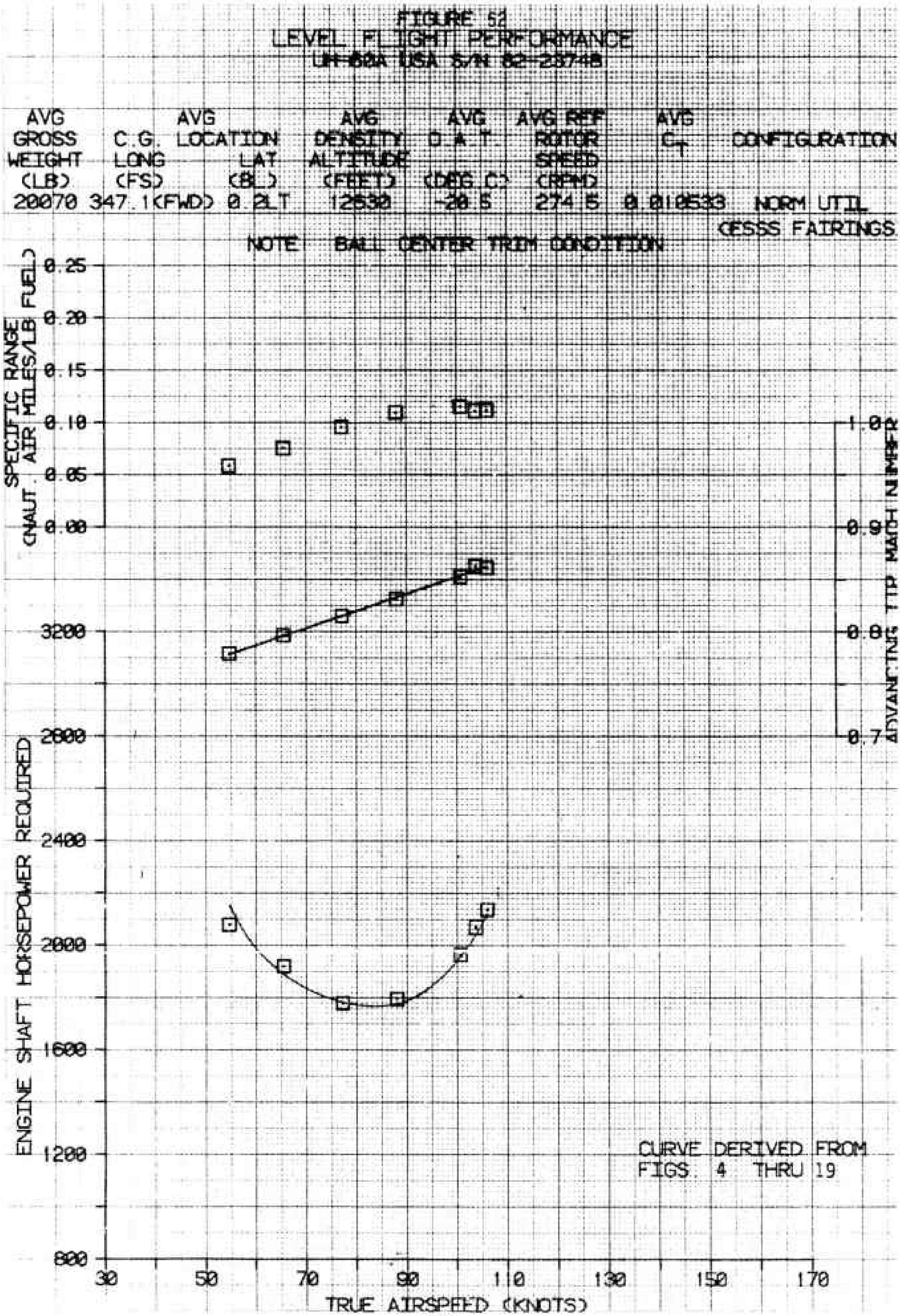


FIGURE 52
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23748
AVG GROSS WEIGHT (LB) 20070
AVG C.G. LOCATION LONG (FS) 347.1(FWD) LAT (BL) 0.2LT
AVG DENSITY ALTITUDE (FEET) 12530
AVG O.A.T. (DEG C) -20.5
AVG REF ROTOR SPEED (RPM) 274.5
AVG C_T 0.010533
CONFIGURATION NORM UTIL (ESSS FAIRINGS
NOTE: BALL CENTER TRIM CONDITION
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
ADVANCING TIP MACH NUMBER
1.0
0.9
0.8
0.7
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
CURVE DERIVED FROM
FIGS. 4 THRU 19
30
50
70
90
110
130
150
170
TRUE AIRSPEED (KNOTS)

FIGURE 53
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 82-23746

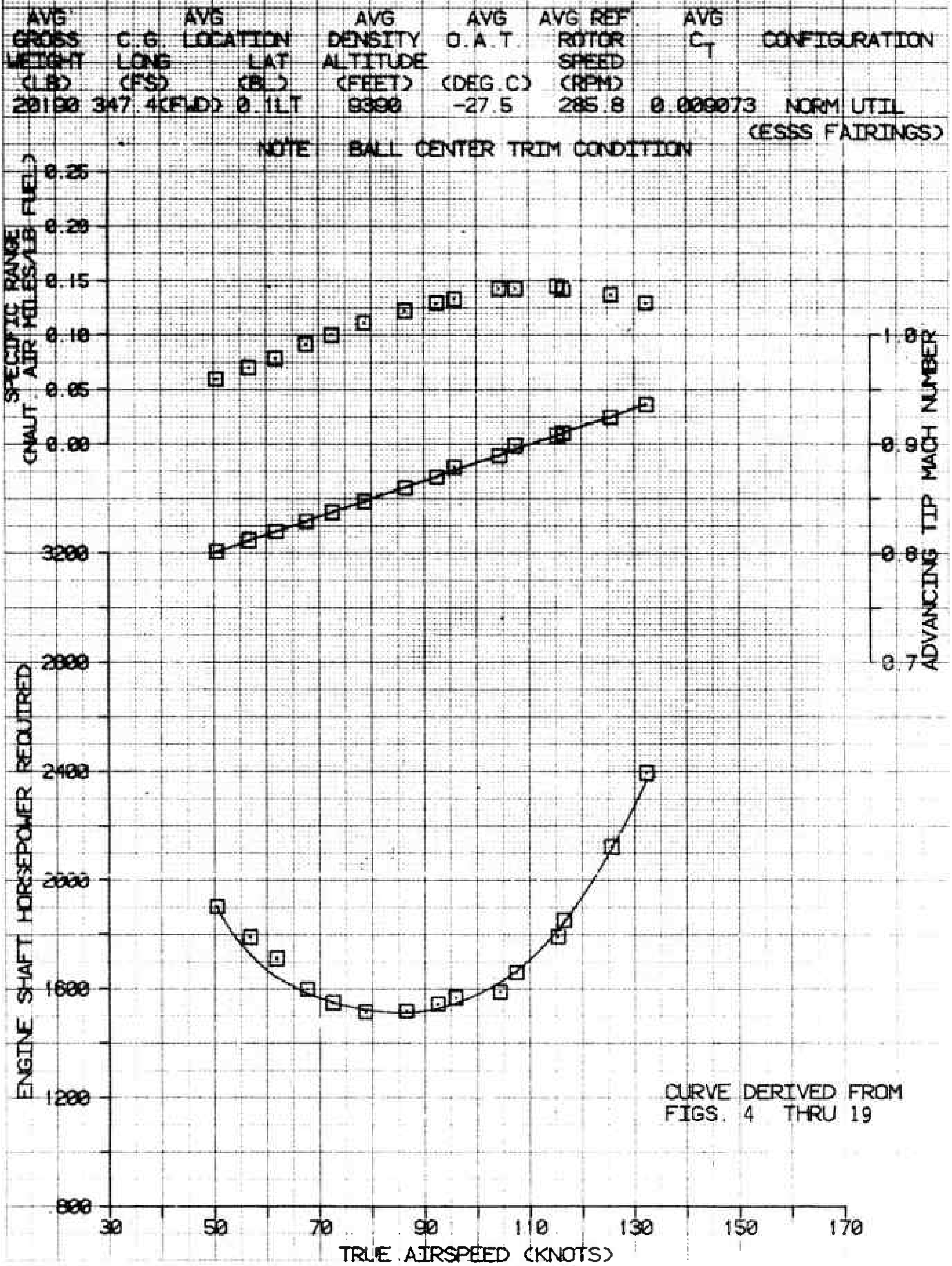
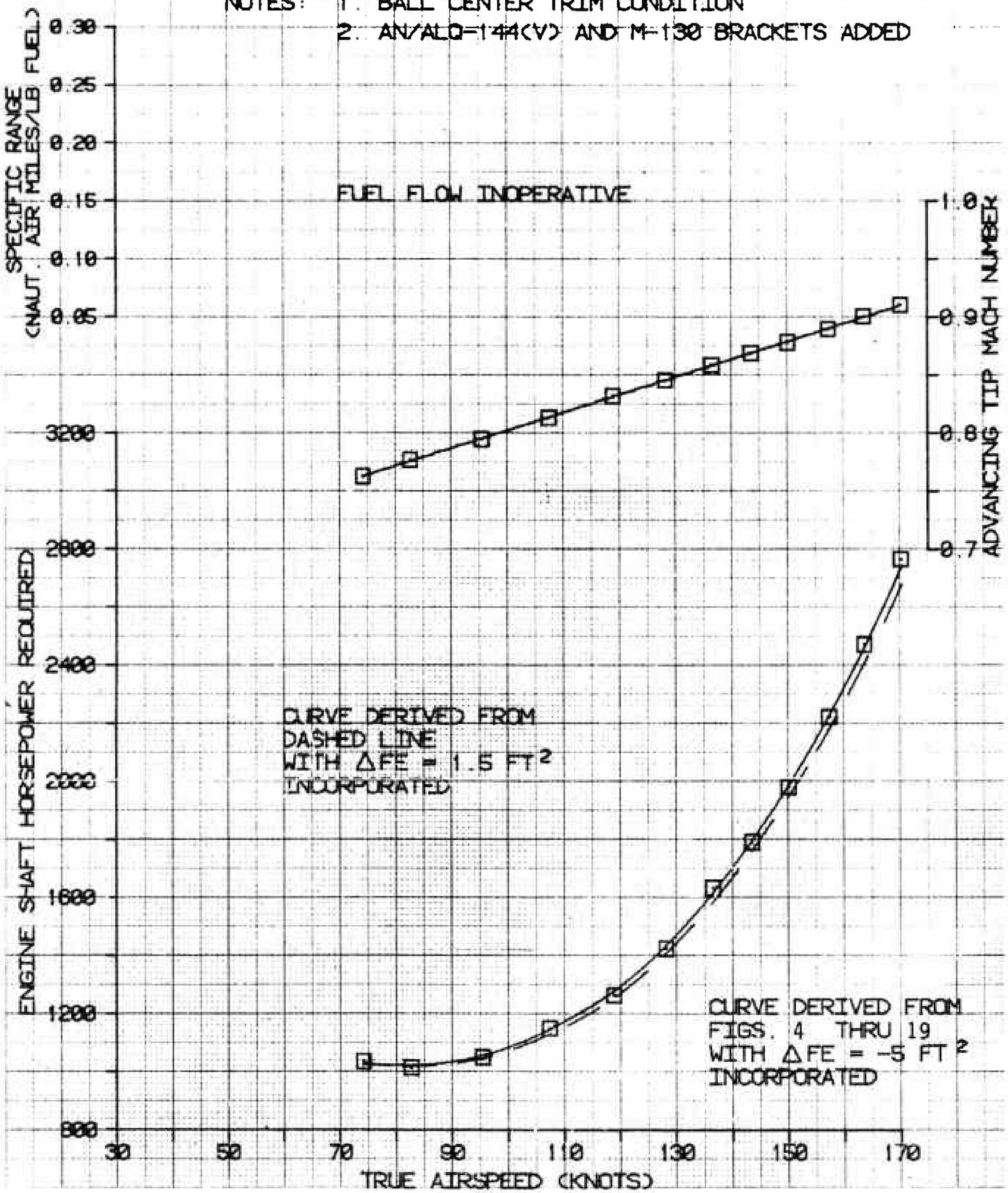


FIGURE 54
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15950	348.1	0.0	6580	9.5	258.3	0.006991	NORM UTIL (NOTE 2.)

NOTES: 1. BALL CENTER TRIM CONDITION
2. AN/ALQ-144(V) AND M-130 BRACKETS ADDED



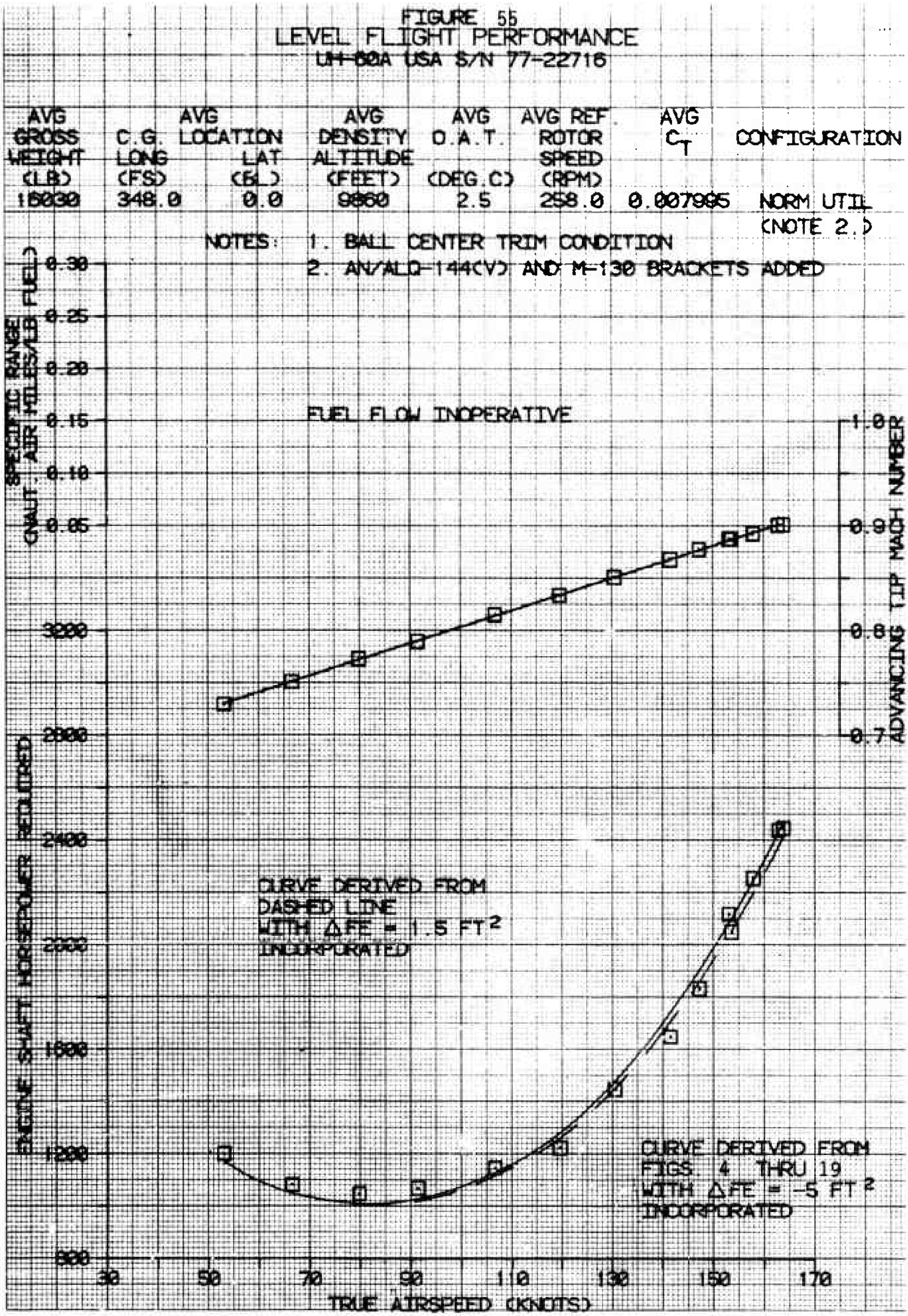


FIGURE 55
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716
AVG GROSS WEIGHT (LB) 16030
AVG C.G. LOCATION LONG (FS) 348.0
LAT (BL) 0.0
AVG DENSITY ALTITUDE (FEET) 9860
AVG O.A.T. (DEG.C) 2.5
AVG REF ROTOR SPEED (RPM) 258.0
AVG C_T 0.007995
CONFIGURATION NORM UTIL (NOTE 2.)
NOTES: 1. BALL CENTER TRIM CONDITION
2. AN/ALQ-144(V) AND M-130 BRACKETS ADDED
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.30
0.25
0.20
0.15
0.10
0.05
FUEL FLOW INOPERATIVE
ADVANCING TIP MACH NUMBER
1.0
0.9
0.8
0.7
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
CURVE DERIVED FROM DASHED LINE WITH ΔFE = 1.5 FT²
INCORPORATED
CURVE DERIVED FROM FIGS. 4 THRU 19
WITH ΔFE = -5 FT²
INCORPORATED
30
50
70
90
110
130
150
170
TRUE AIRSPEED (KNOTS)

FIGURE 56
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15840	347.1	0.0	14000	3.0	258.8	0.008943	NORM UTIL (NOTE 2.)

NOTES: 1. BALL CENTER TRIM CONDITION
2. AN/ALQ-144(V) AND M-130 BRACKETS ADDED

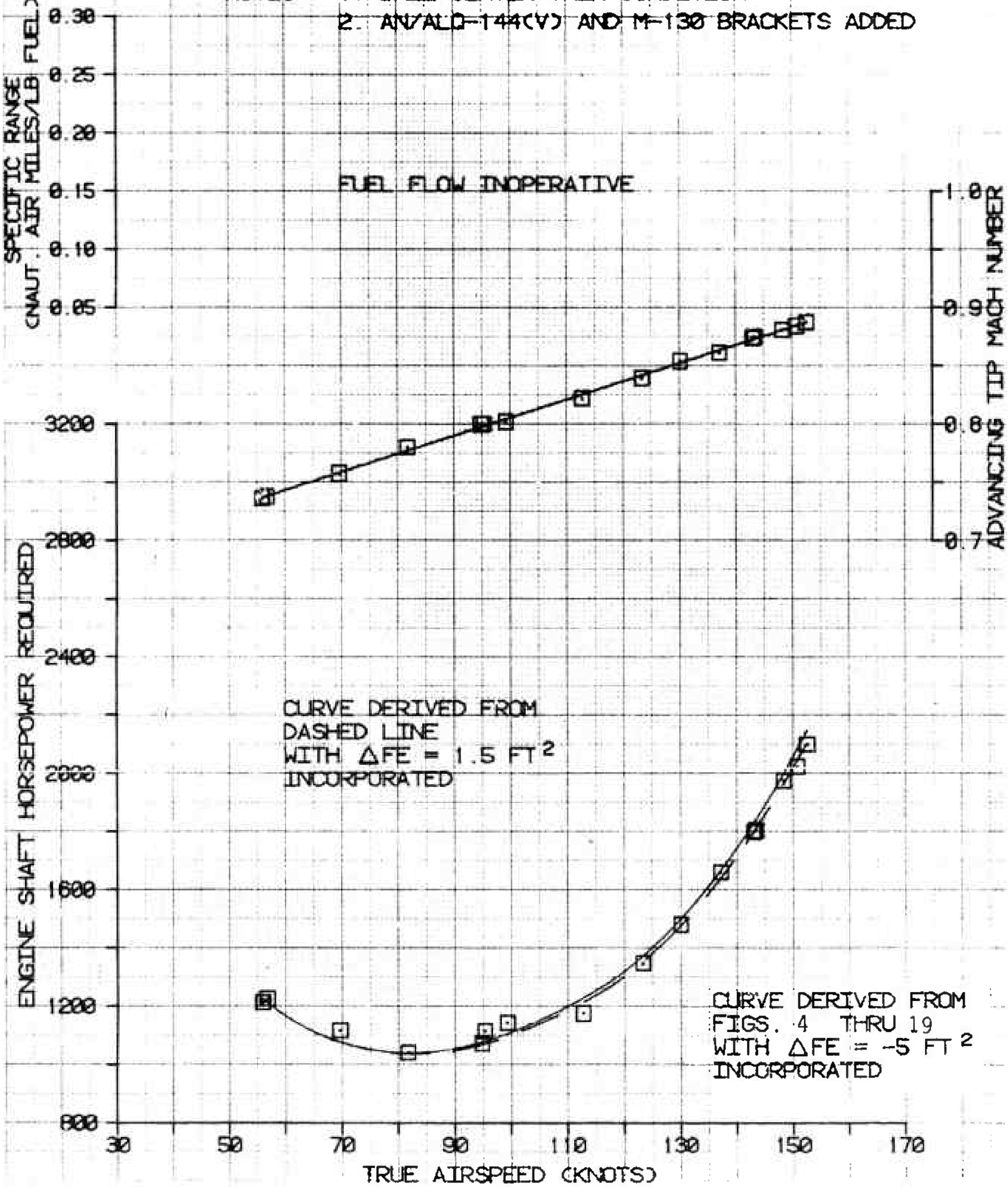
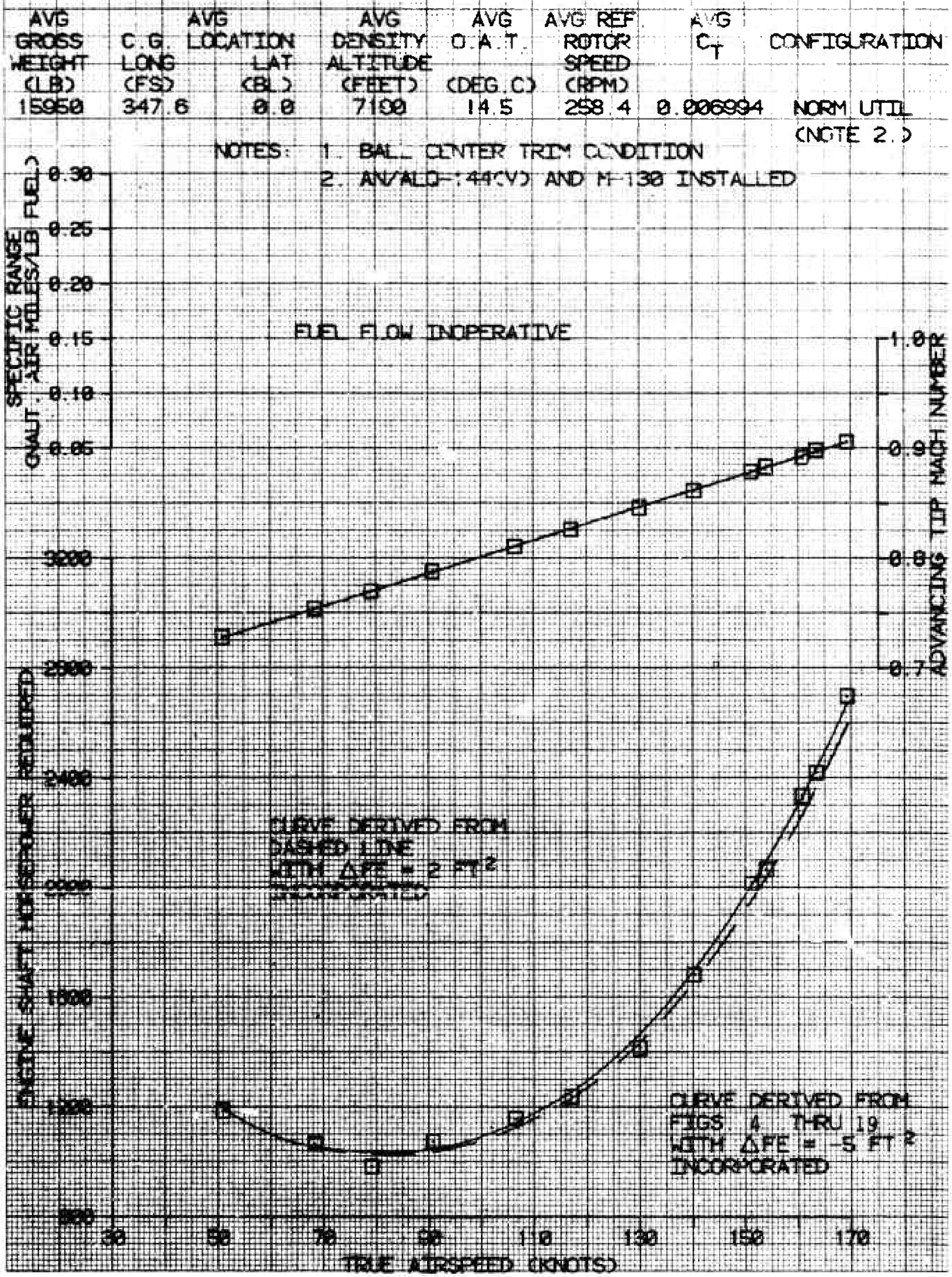


FIGURE 57
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716



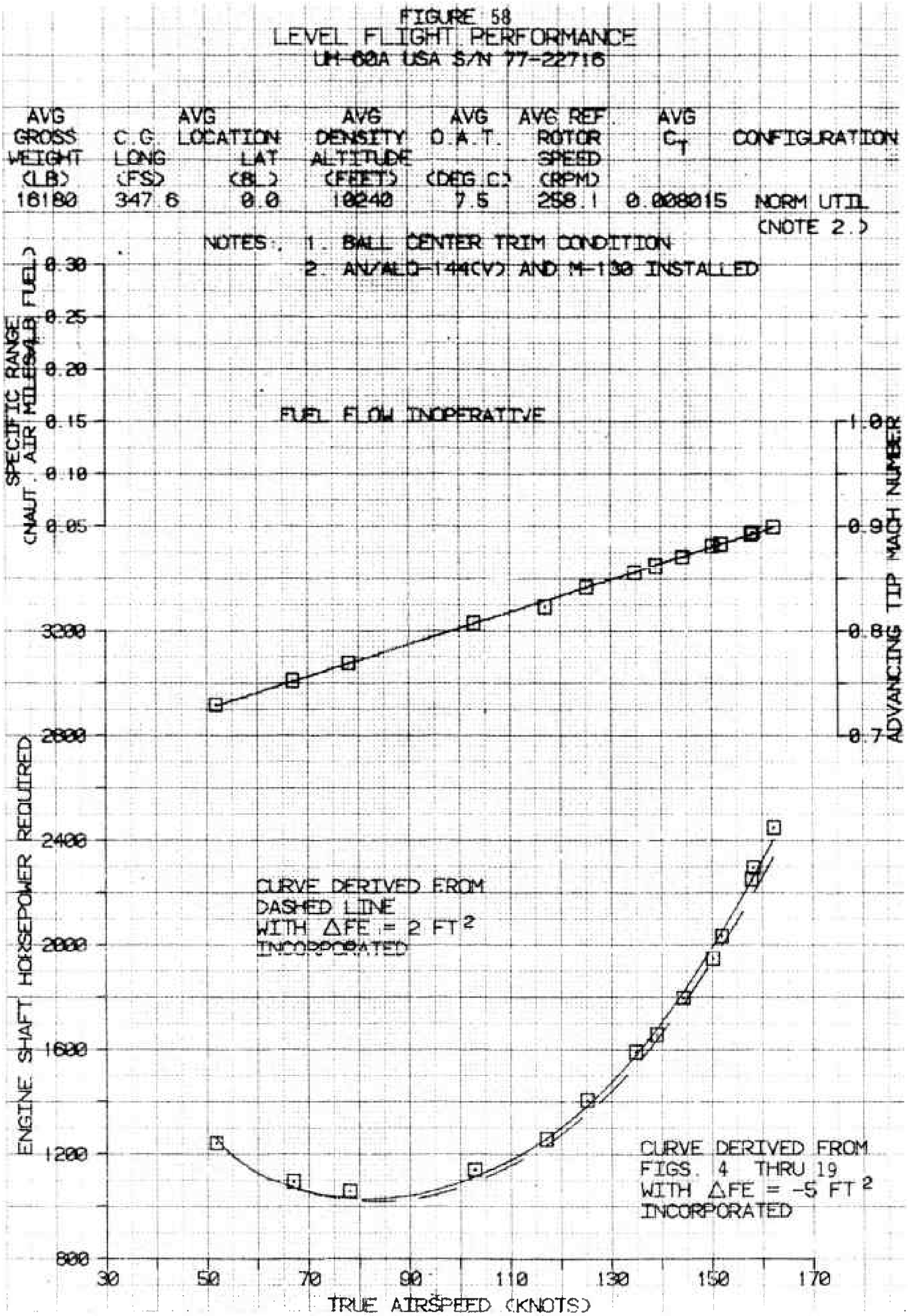


FIGURE 58
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716
AVG GROSS WEIGHT (LB) 16180
AVG C.G. LOCATION LONG (FS) 347.6 LAT (BL) 0.0
AVG DENSITY ALTITUDE (FEET) 10240
AVG O.A.T. (DEG C) 7.5
AVG REF. ROTOR SPEED (RPM) 258.1
AVG C_T 0.008015
CONFIGURATION NORM UTIL (NOTE 2.)
NOTES: 1. BALL CENTER TRIM CONDITION
2. AN/ALQ-144(V) AND M-130 INSTALLED
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.30
0.25
0.20
0.15
0.10
0.05
FUEL FLOW INOPERATIVE
ADVANCING TIP MACH NUMBER
1.0
0.9
0.8
0.7
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
CURVE DERIVED FROM DASHED LINE WITH ΔFE = 2 FT²
INCORPORATED
CURVE DERIVED FROM FIGS. 4 THRU 19 WITH ΔFE = -5 FT²
INCORPORATED
30
50
70
90
110
130
150
170
TRUE AIRSPEED (KNOTS)

FIGURE 59
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG O.A.T. (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
16110	347.4	0.0	13470	2.0	258.2	0.009015	NORM UTIL (NOTE 2.)

NOTES: 1. BALL CENTER TRIM CONDITION
2. AN/ALQ-144(V) AND M-130 INSTALLED

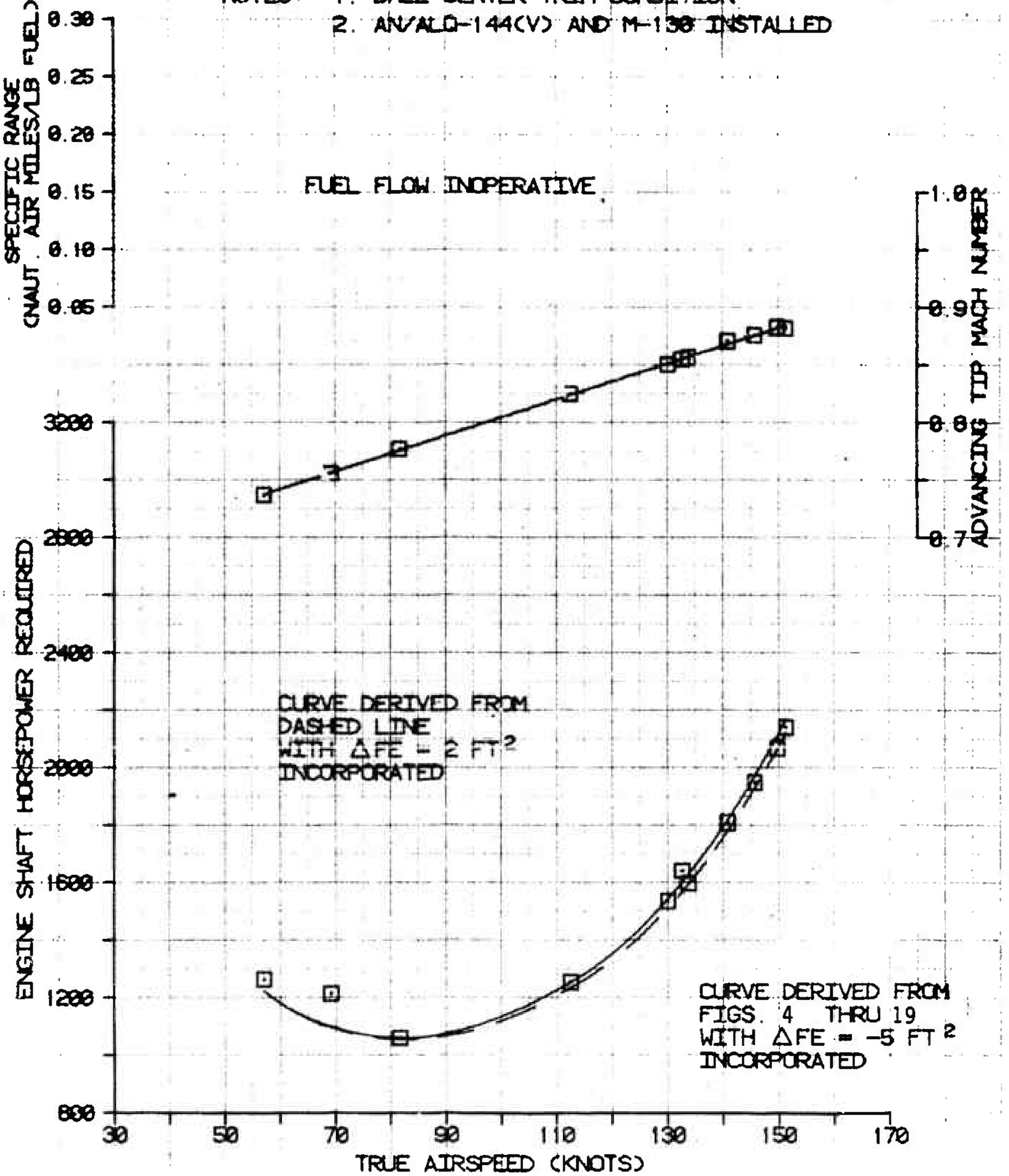


FIGURE 69

INHERENT SIDESLIP

UH-60A USA S/N 77-22716 and S/N 82-23748

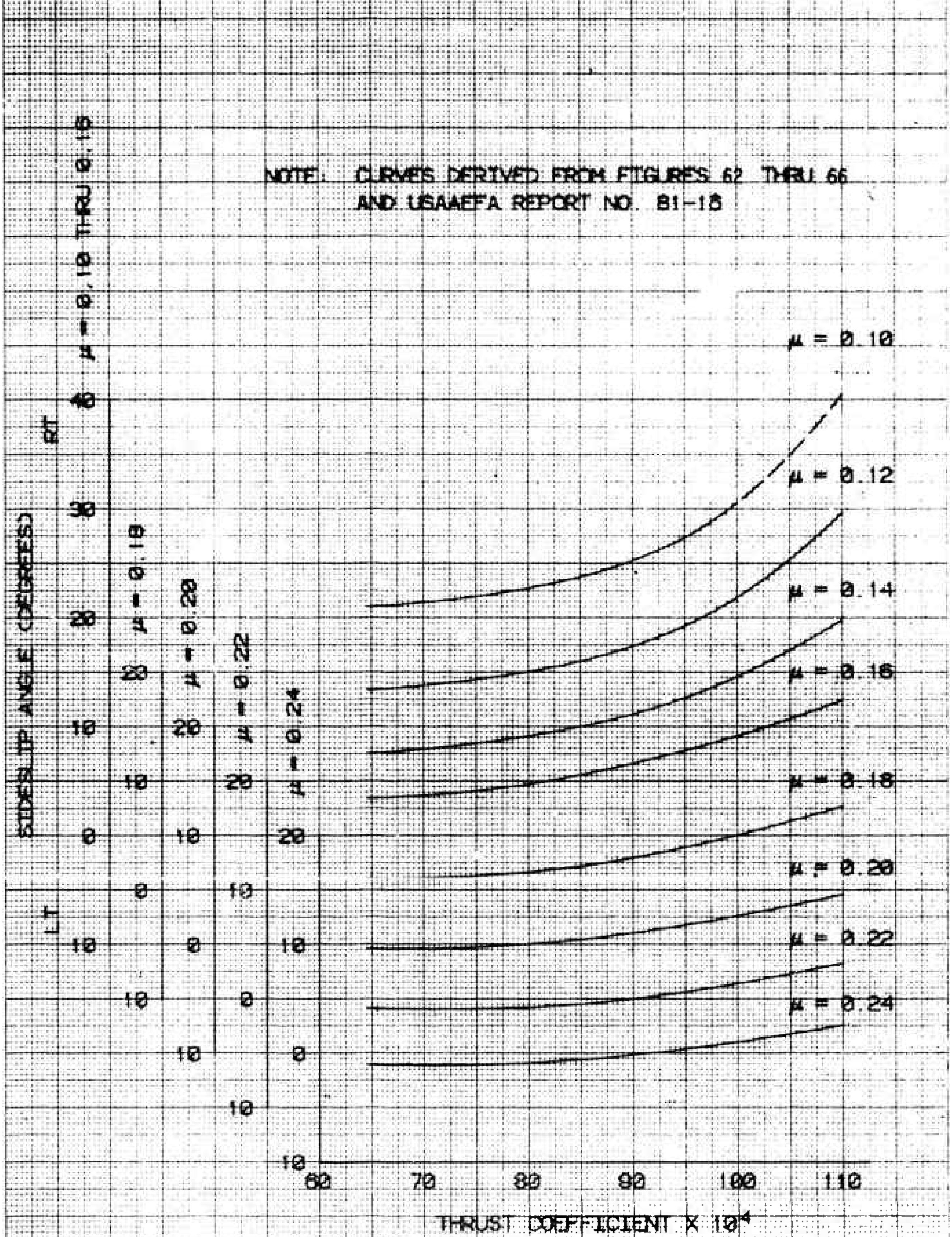


FIGURE 51

INHERENT SIDESLIP

UH-60A USA S/N 77-22716 and S/N 82-23748

NOTE: CURVES DERIVED FROM FIGURES 62 THRU 66
AND USAAEFA REPORT NO. 81-16

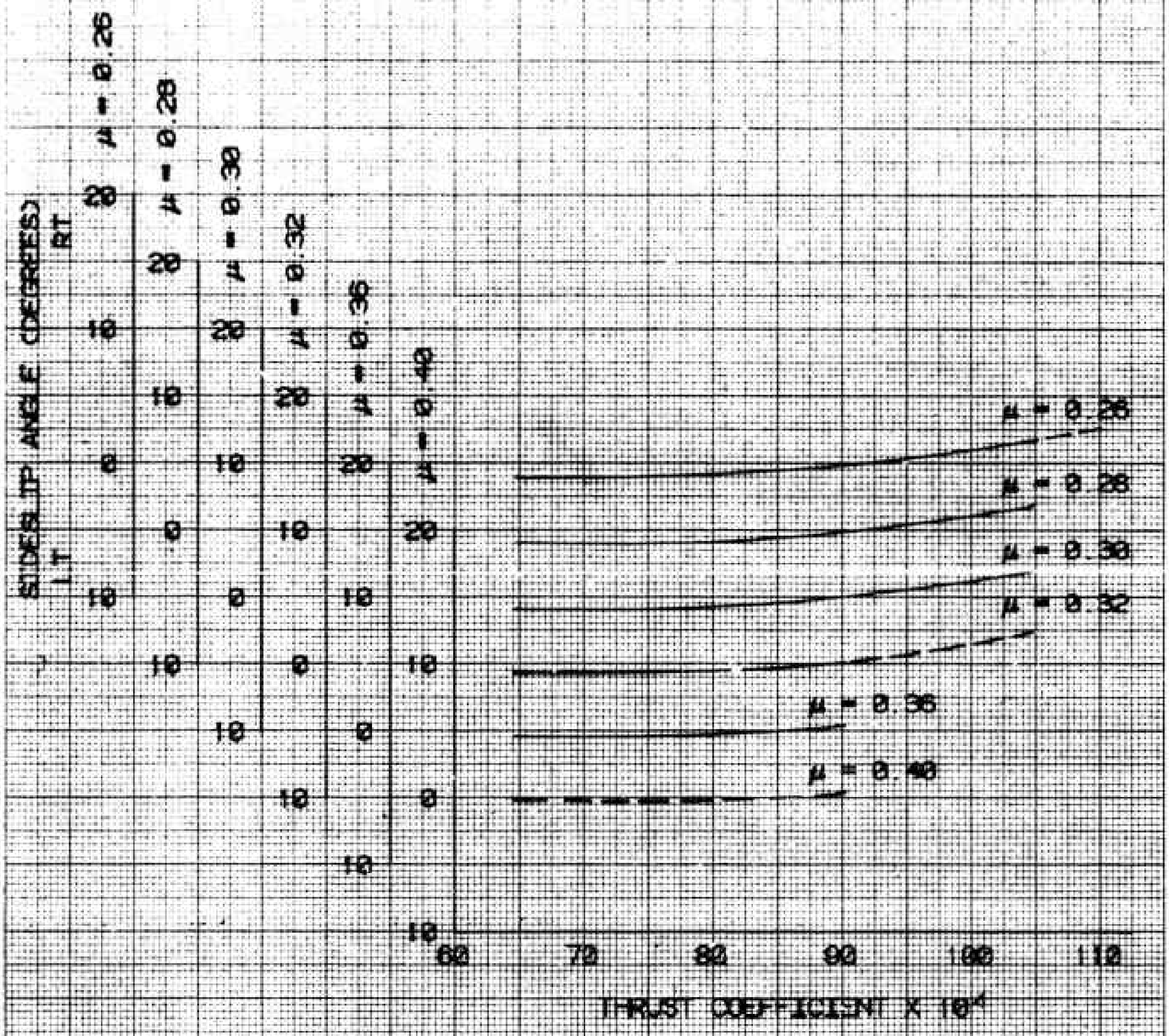


FIGURE 62

SIDESLIP IN BALL CENTER LEVEL FLIGHT

UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG REFERRED ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT
△	21690	347.3(FWD)	0.1LT	8870	16.0	258.1	0.009981
◇	18190	347.1	0.1LT	13840	7.0	258.0	0.009019
★	15920	347.1	0.1LT	11520	14.5	258.2	0.008206
+	14470	347.0	0.1LT	10290	14.5	258.0	0.007010

NOTES: 1. NORMAL UTILITY CONFIGURATION.
2. CURVES OBTAINED FROM FIGURES 60 AND 61.

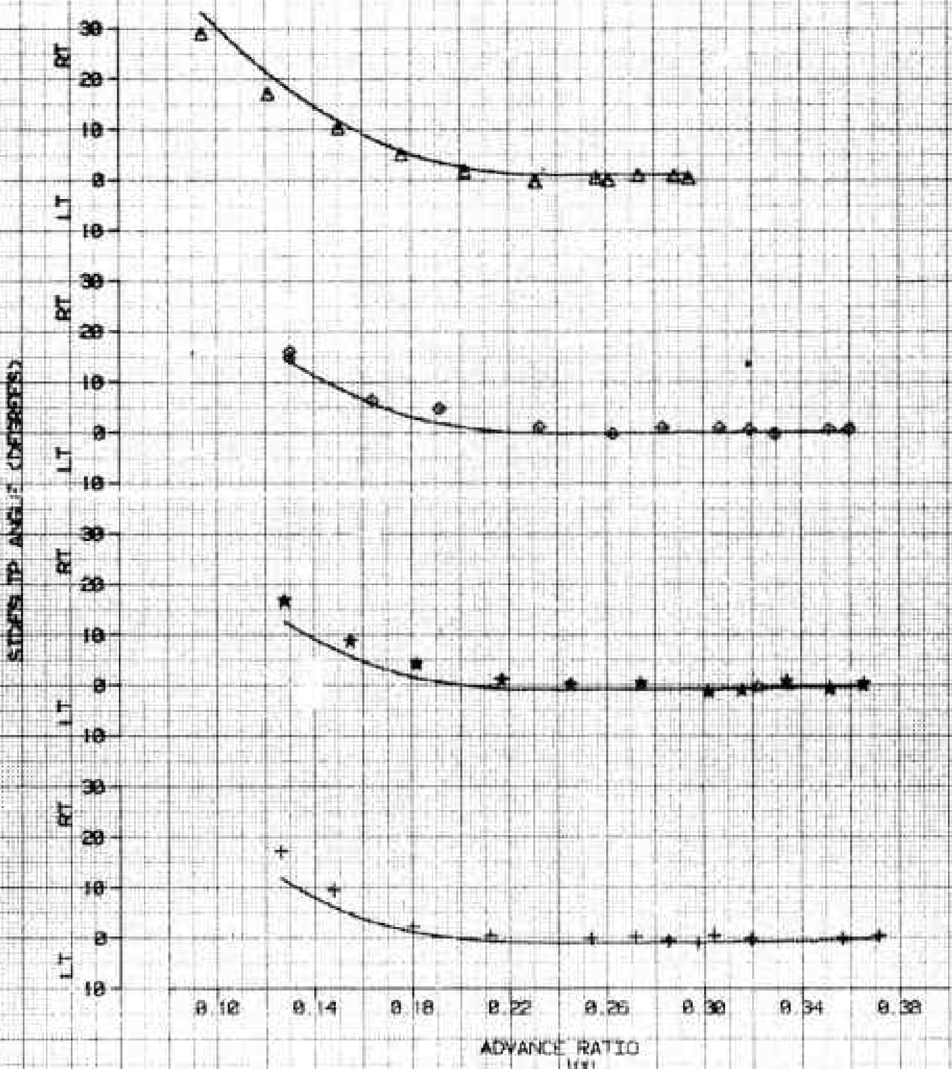


FIGURE 63

SIDESLIP IN BALL CENTER LEVEL FLIGHT

UH-60A USA S/N 82-23748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG REFERRED ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT
△	10870	347.2(FWD)	0.3LT	12370	0.5	259.0	0.010357
◇	10730	347.8(FWD)	0.3LT	7070	11.0	257.5	0.009835
★	10720	347.5(FWD)	0.3LT	4790	17.0	257.0	0.009005
+	14870	347.0	0.3LT	7150	16.0	258.5	0.006474

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. CURVES OBTAINED FROM FIGURES 60 AND 61

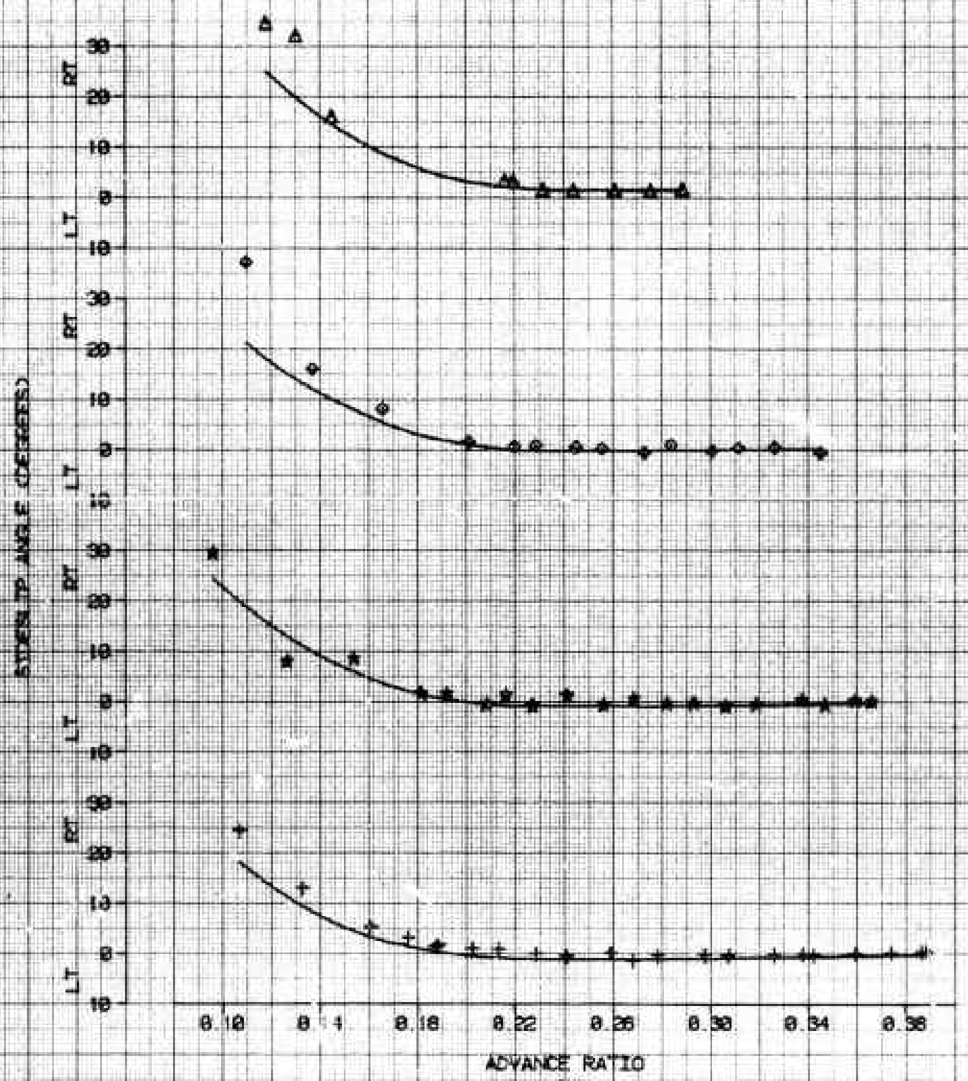


FIGURE 64

SIDESLIP IN BALL CENTER LEVEL FLIGHT

UH-60A USA S/N 82-23748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG CAT (DEG C)	AVG REFERRED ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT
□	19360	347.3(FWD)	0.3LT	10830	16.5	245.4	0.010472
○	19490	347.1(FWD)	0.2LT	6300	23.0	245.1	0.008978
△	15110	346.9	0.4LT	4970	23.5	244.0	0.006510
◇	19600	347.5	0.2LT	13880	3.5	265.5	0.010454
★	19890	347.2(FWD)	0.3LT	9050	8.0	264.5	0.009022
+	17000	347.5	0.1LT	3430	-13.0	264.9	0.006997

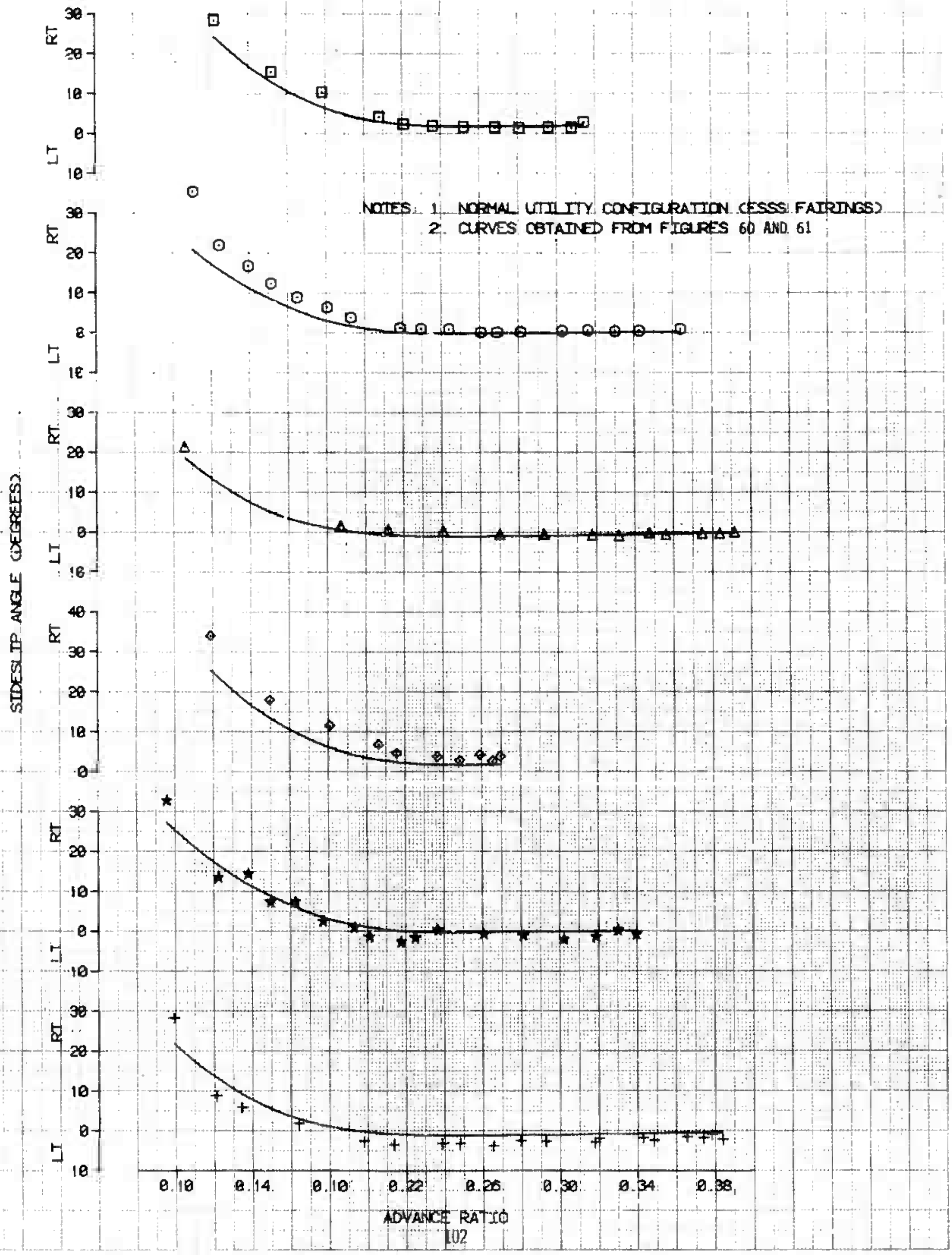


FIGURE 65

SIDESLIP IN BALL CENTER LEVEL FLIGHT

UH-60A USA S/N 82-23748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG REFERRED ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT
⊡	20070	347.1(FWD)	0.2LT	12630	-20.5	274.5	0.010533
○	20180	347.5(FWD)	0.1LT	10200	-30.0	275.7	0.010089
△	20020	347.4(FWD)	0.2LT	8370	-18.5	275.0	0.009010
⌽	19700	347.7(FWD)	0.1LT	4520	-23.0	274.9	0.008115
★	17000	347.2	0.1LT	5000	-16.0	275.3	0.006922
+	20190	347.4(FWD)	0.1LT	9300	-27.5	285.6	0.009073

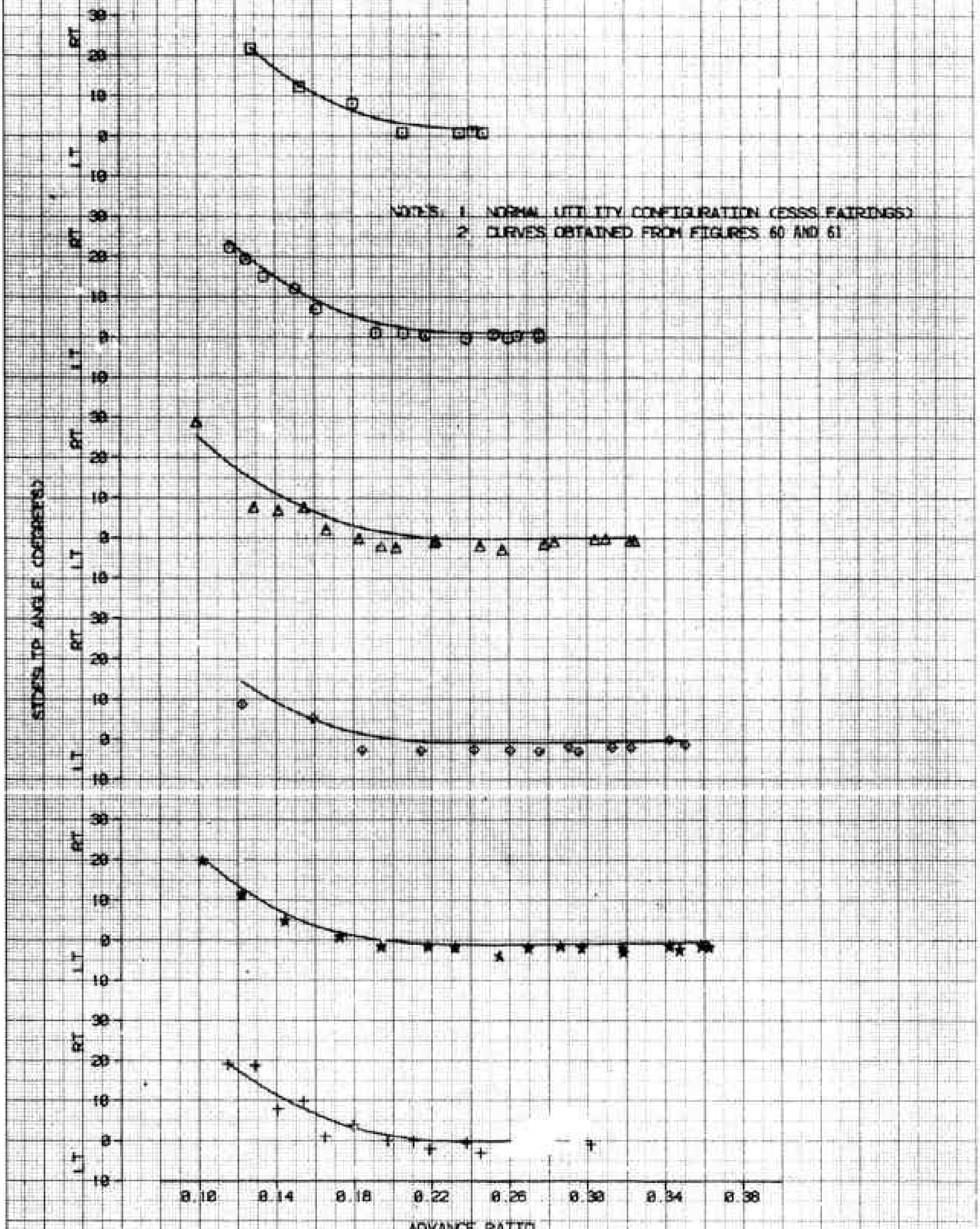


FIGURE 66

SIDESLIP IN BALL CENTER LEVEL FLIGHT

UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG REFERRED ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT
⊡	16110	347.4	0.0	13470	2.0	258.2	0.009015
⊙	18180	347.6	0.0	10240	7.5	258.1	0.008015
△	15950	347.6	0.0	7190	14.5	258.4	0.008994
◇	15840	347.1	0.0	14000	3.0	258.8	0.008943
★	16030	348.0	0.0	9860	2.5	258.0	0.007995
+	15950	348.1	0.0	6580	9.5	258.3	0.006991

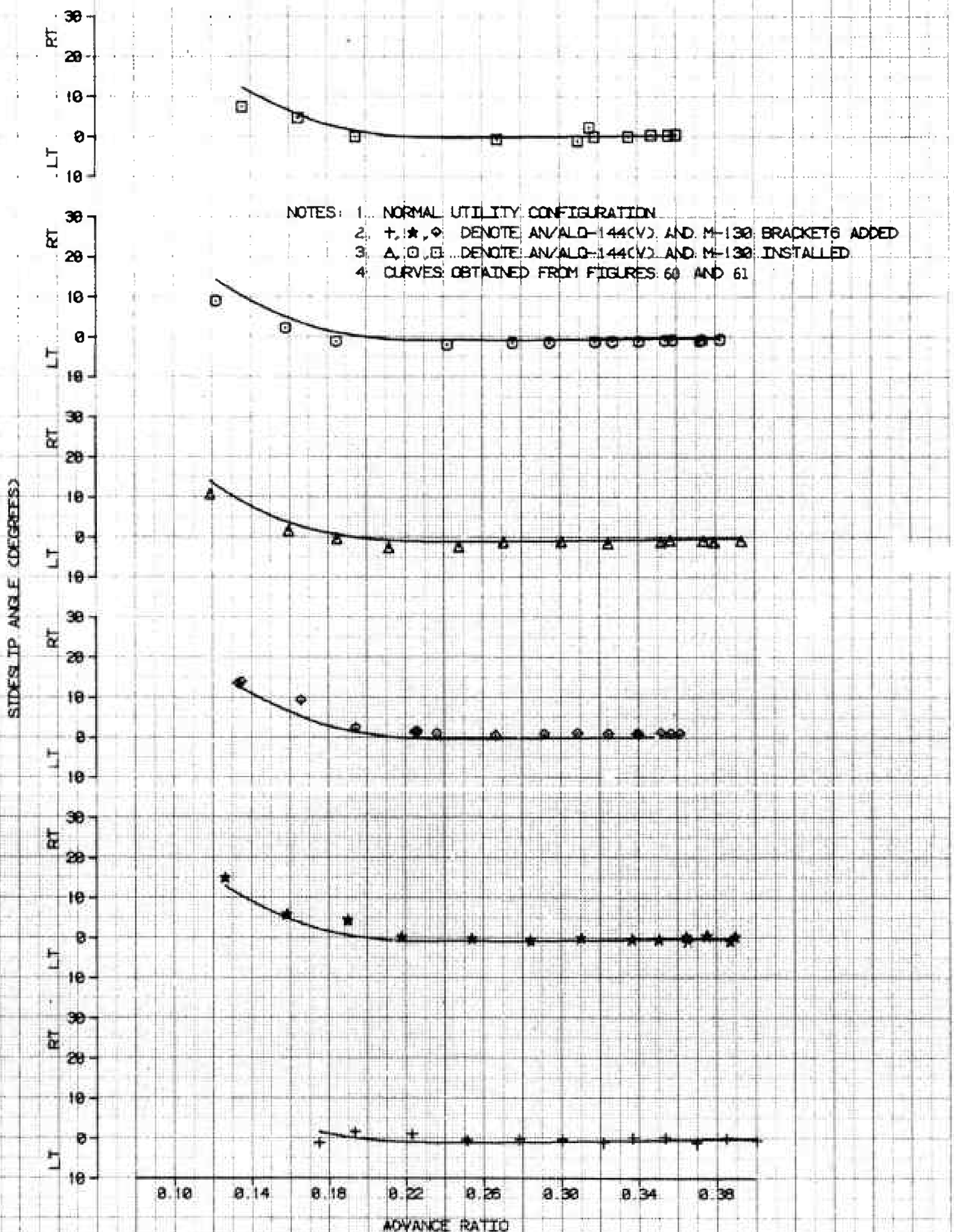


FIGURE 67

CHANGE IN EQUIVALENT FLAT PLATE AREA vs SIDESLIP

UH-60A USA S/N 77-22718 and S/N 82-23748

SYMBOL	AVG TRUE AIRSPEED (KT)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG REFERRED ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
□	100	347.5	0.4LT	258	0.008499	NORM UTIL (ESSS FAIRINGS)
⊙	120	347.3	0.3LT	257	0.008554	NORM UTIL (ESSS FAIRINGS)
△	145	347.3	0.3LT	258	0.008529	NORM UTIL (ESSS FAIRINGS)
+	157	346.9	0.3LT	257	0.008542	NORM UTIL (ESSS FAIRINGS)
◇	98	347.1	0.2LT	258	0.009013	NORM UTIL
⊠	100	347.3	0.3LT	258	0.009014	NORM UTIL (ESSS FAIRINGS)
⌧	104	347.6	0.1LT	285	0.009132	NORM UTIL (ESSS FAIRINGS)
⊗	117	347.3	0.0	258	0.008882	NORM UTIL
⊞	124	347.3	0.3LT	258	0.008846	NORM UTIL (ESSS FAIRINGS)
▽	140	347.0	0.3LT	257	0.008941	NORM UTIL (ESSS FAIRINGS)
★	140	347.2	0.0	258	0.009070	NORM UTIL
*	78	347.1	0.3LT	257	0.010572	NORM UTIL (ESSS FAIRINGS)
×	85	347.4	0.2LT	275	0.010575	NORM UTIL (ESSS FAIRINGS)
↑	104	347.2	0.3LT	258	0.010581	NORM UTIL (ESSS FAIRINGS)
↓	75	347.0	0.0	259	0.011093	NORM UTIL
•	98	347.0	0.0	258	0.011115	NORM UTIL

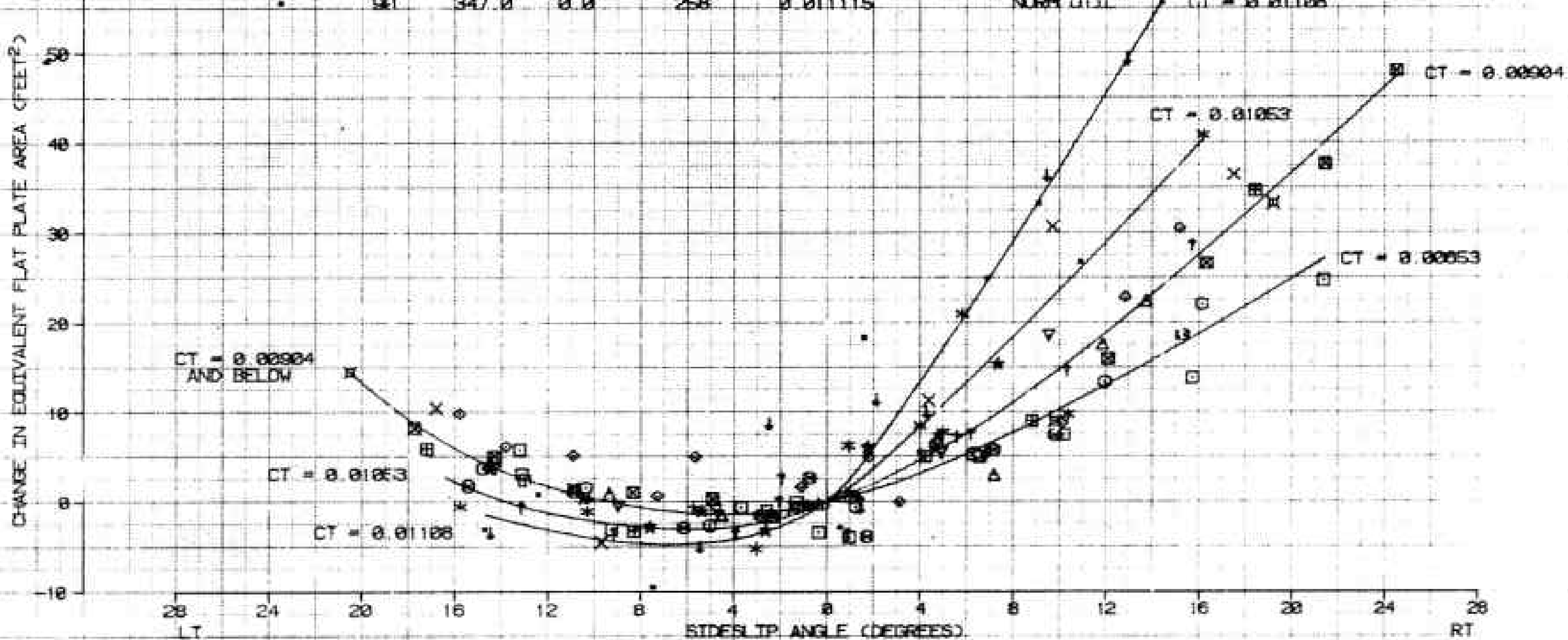


FIGURE 68

EFFECT OF STABILATOR POSITION ON LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 82-23748

$C_T = 0.0090$

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTERED TRIM CONDITION
3. REFERRED ROTOR SPEED = 258
4. TEUP STABILATOR MOVEMENT PRODUCES AN INCREASE IN POWER REQUIRED
5. TEDN STABILATOR MOVEMENT PRODUCES A DECREASE IN POWER REQUIRED
6. CURVES OBTAINED FROM FIGURE 69

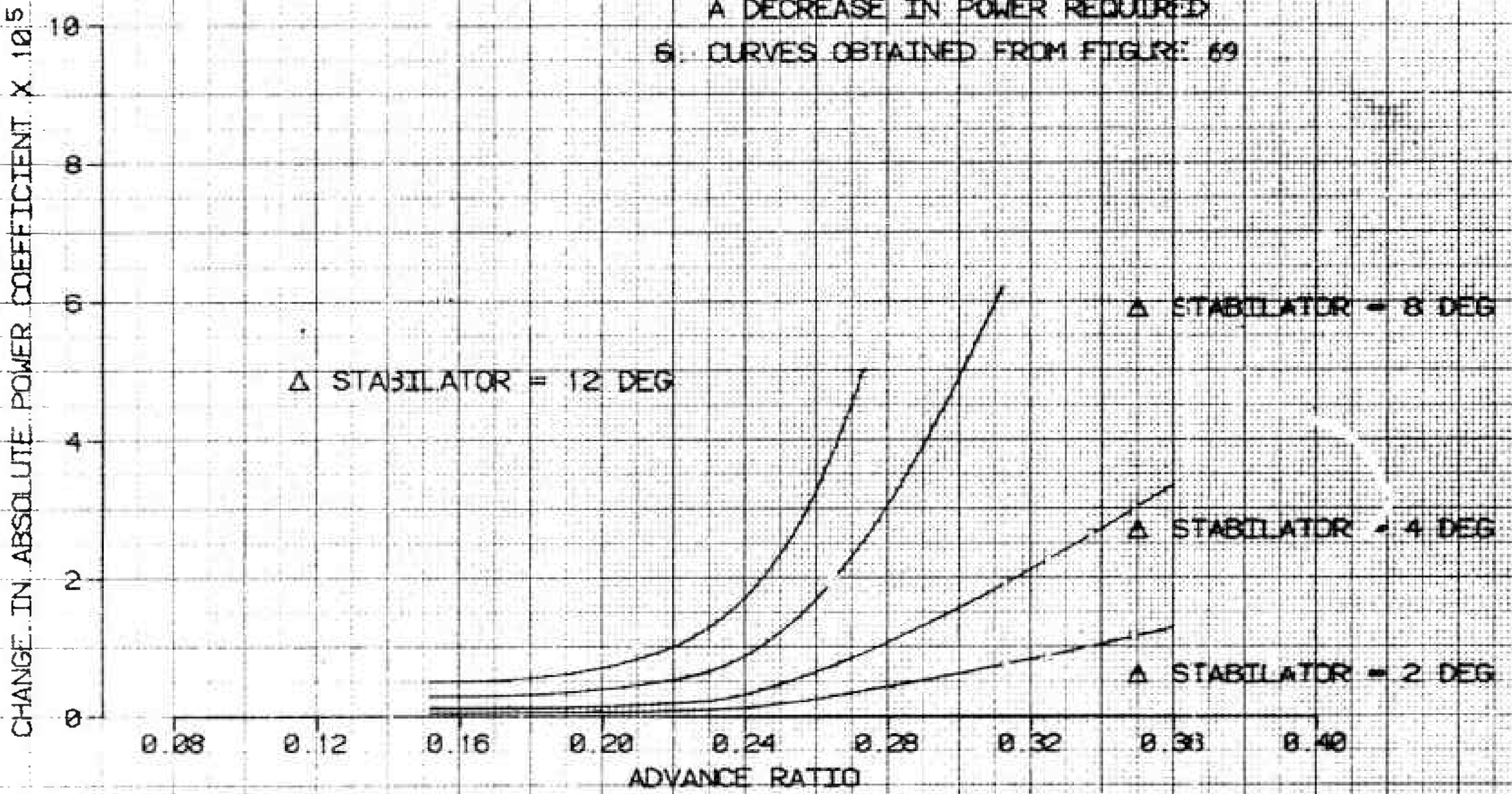


FIGURE 69

CHANGE IN POWER COEFFICIENT WITH STABILATOR POSITION

UH-60A USA S/N 77-22716

$C_T = 0.0090$

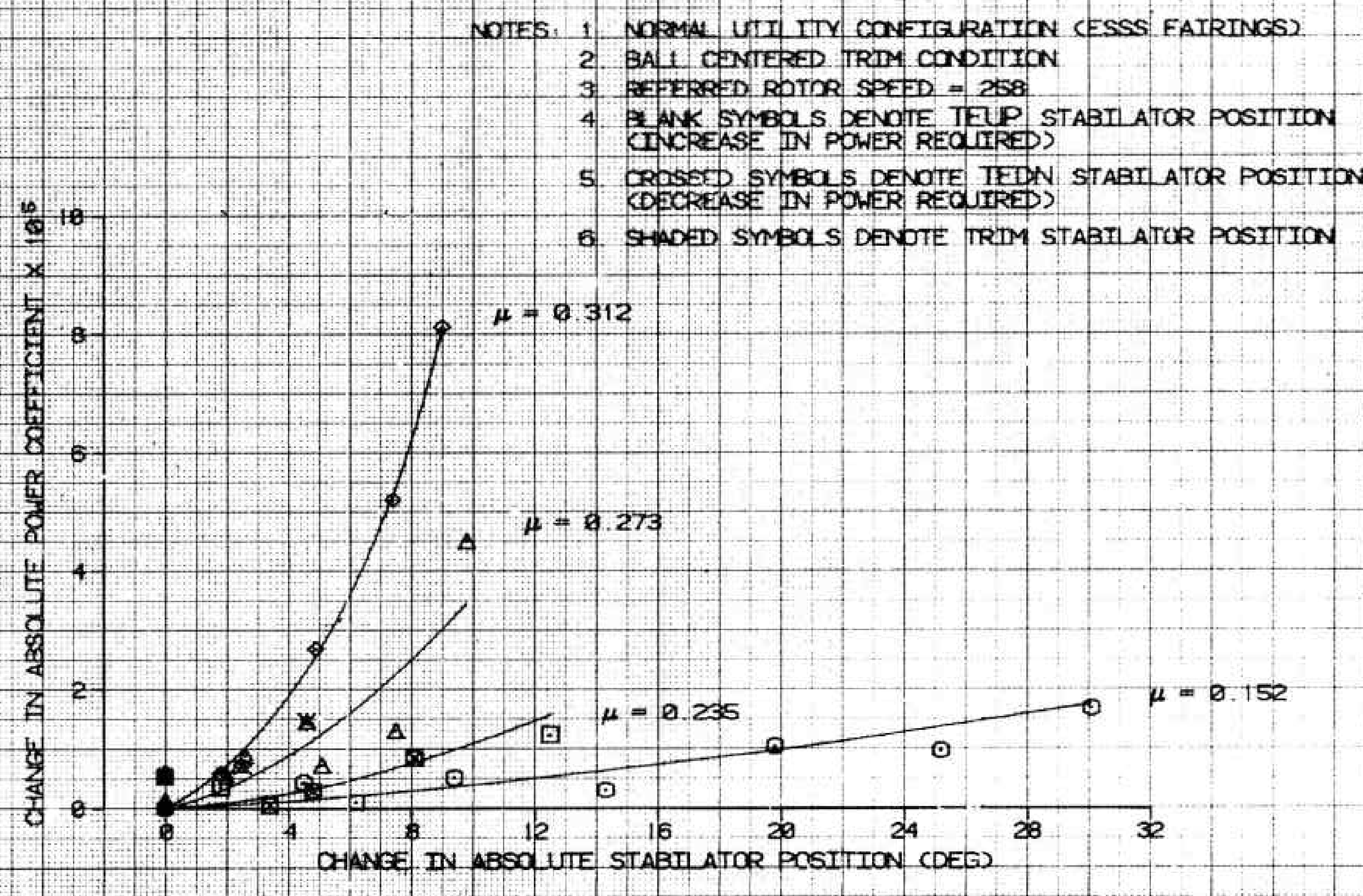


FIGURE 70

CHANGE IN POWER COEFFICIENT WITH STABILATOR POSITION

UH-60A USA S/N 82-23748

μ = 0.245

NOTES: 1. NORMAL UTILITY CONFIGURATION (ESSS FAIRINGS)
2. BALL CENTERED TRIM CONDITION
3. REFERRED ROTOR SPEED = 258
4. BLANK SYMBOLS DENOTE TEUP STABILATOR POSITION (INCREASE IN POWER REQUIRED)
5. CROSSED SYMBOLS DENOTE TEDN STABILATOR POSITION (DECREASE IN POWER REQUIRED)
6. SHADED SYMBOL DENOTES TRIM STABILATOR POSITION

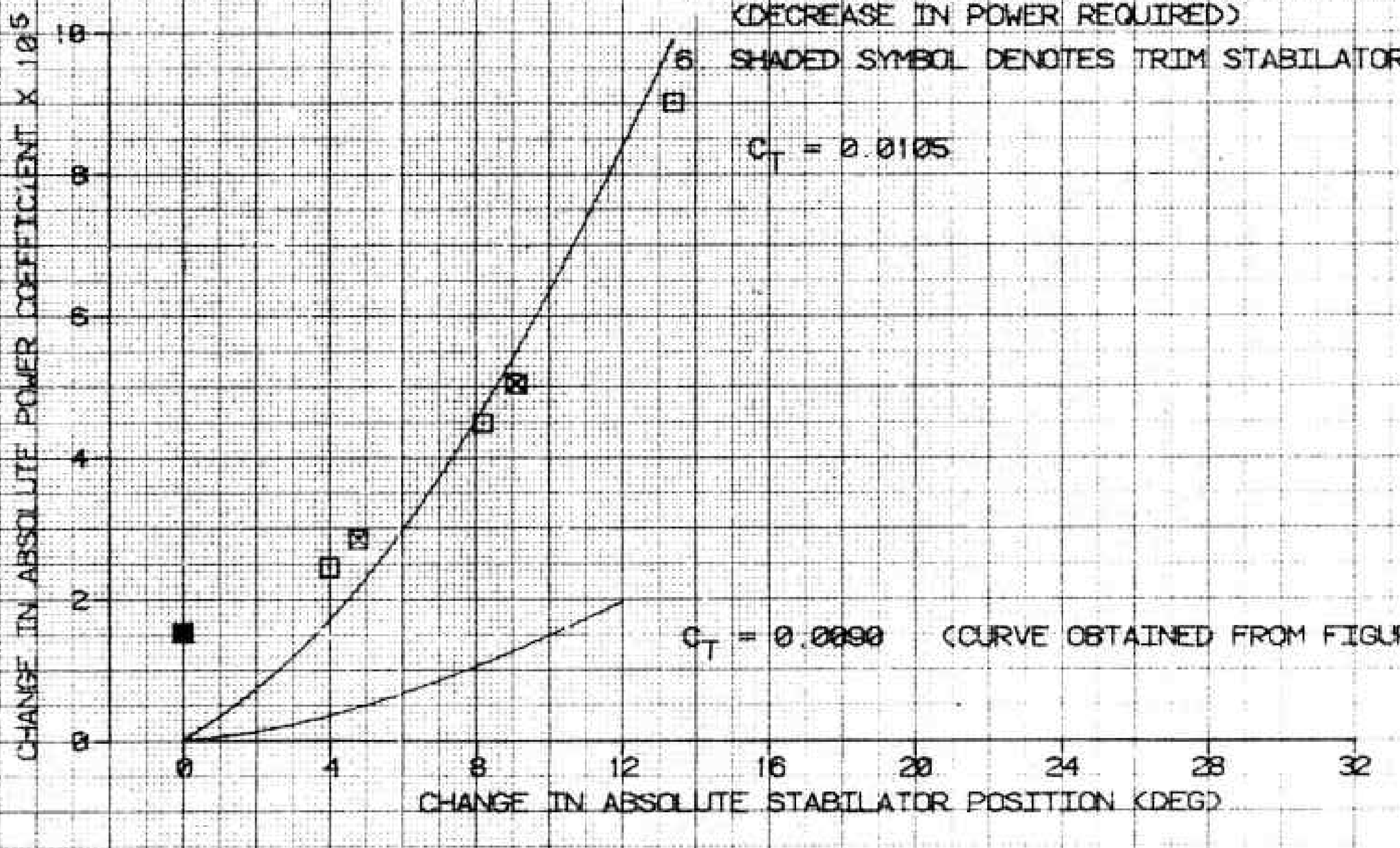
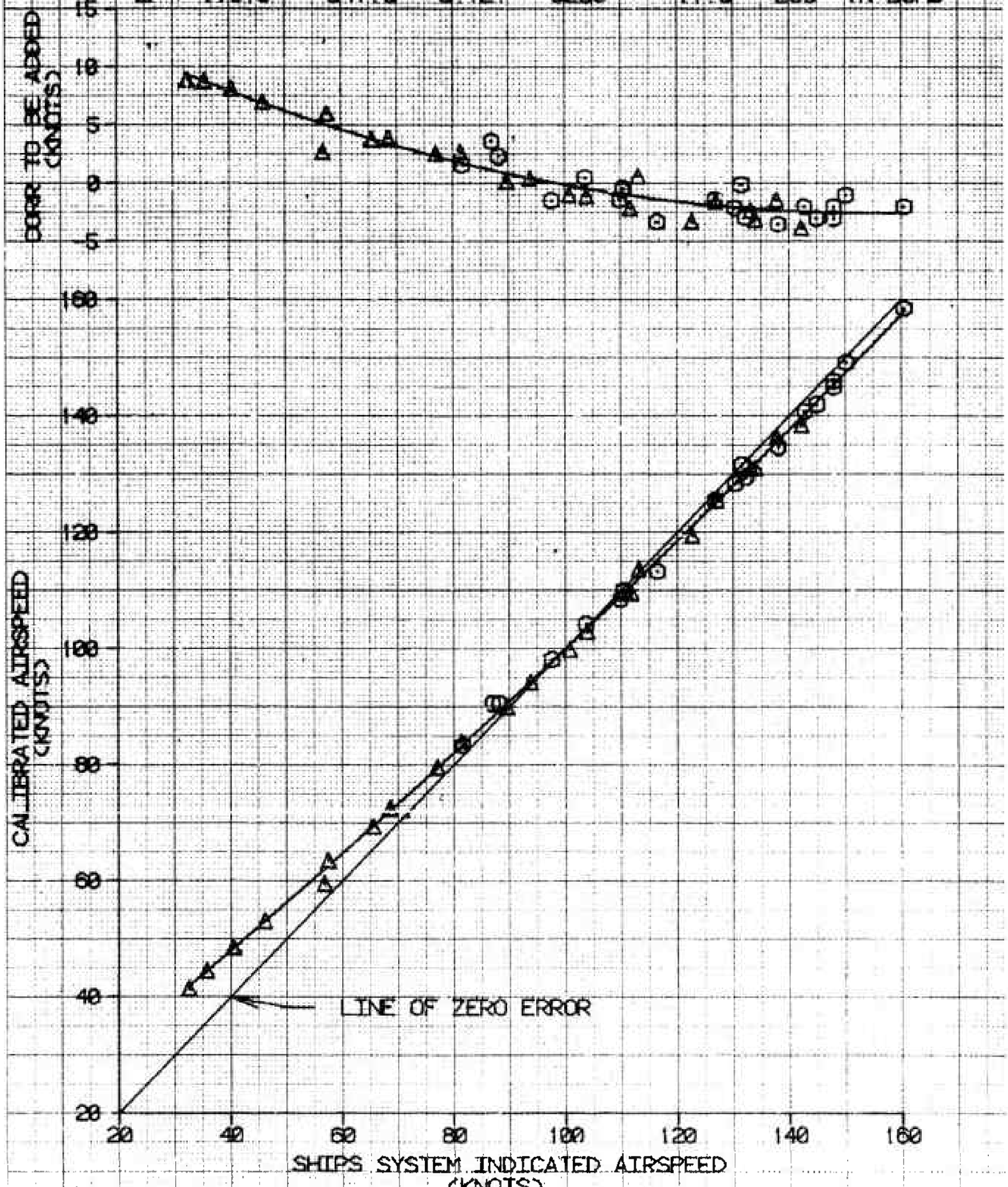


FIGURE 71

SHIP SYSTEM AIRSPEED CALIBRATION IN LEVEL FLIGHT

UH-60A USA S/N 82-23748

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TEST METHOD
⊡	16920	346.4(FWD)	0.2 LT	6880	5.5	257	T28 PACE
△	17310	347.8	0.1LT	3290	-11.5	258	TR BOMB



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