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AEFA Project No. 87-32

AEFA

BASELINE PERFORMANCE VERIFICATION OF THE 12TH YEAR PRODUCTION UH-60A BLACK HAWK HELICOPTER

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January 1989

Final Report

AVIATION ENGINEERING FLIGHT ACTIVITY
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REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED	**1b. RESTRICTIVE MARKINGS**
2a. SECURITY CLASSIFICATION AUTHORITY U.S. ARMY AVIATION SYSTEMS COMMAND	**3. DISTRIBUTION/AVAILABILITY OF REPORT**
2b. DECLASSIFICATION/DOWNGRADING SCHEDULE	
4. PERFORMING ORGANIZATION REPORT NUMBER(S) AEFA PROJECT NO. 87-32	**5. MONITORING ORGANIZATION REPORT NUMBER(S)**
6a. NAME OF PERFORMING ORGANIZATION U.S. ARMY AVIATION ENGINEERING FLIGHT ACTIVITY / **6b. OFFICE SYMBOL** *(If applicable)*	**7a. NAME OF MONITORING ORGANIZATION**
6c. ADDRESS *(City, State, and ZIP Code)* EDWARDS AIR FORCE BASE, CALIFORNIA 93523-5000	**7b. ADDRESS** *(City, State, and ZIP Code)*
8a. NAME OF FUNDING/SPONSORING ORGANIZATION U.S. ARMY AVIATION SYSTEMS COMMAND / **8b. OFFICE SYMBOL** *(If applicable)*	**9. PROCUREMENT INSTRUMENT IDENTIFICATION NUMBER**
8c. ADDRESS *(City, State, and ZIP Code)* 4300 GOODFELLOW BLVD. ST. LOUIS, MO 63120-1798	**10. SOURCE OF FUNDING NUMBERS**

10. SOURCE OF FUNDING NUMBERS

PROGRAM ELEMENT NO.	PROJECT NO.	TASK NO	WORK UNIT ACCESSION NO.
68-8BE433-02	-68-EC		

11. TITLE *(Include Security Classification)*
Baseline Performance Verification of the 12th Year Production UH-60A Black Hawk Helicopter. UNCLASSIFIED

12. PERSONAL AUTHOR(S)
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13a. TYPE OF REPORT	13b. TIME COVERED	14. DATE OF REPORT *(Year, Month, Day)*	15. PAGE COUNT
FINAL	FROM 15/08/88 TO 02/09/88	January 1989	80

16. SUPPLEMENTARY NOTATION

17. COSATI CODES			18. SUBJECT TERMS *(Continue on reverse if necessary and identify by block number)*
FIELD	GROUP	SUB-GROUP	12th Year Production UH-60A Helicopter, Performance Baseline, Prime Item Development Specification (PIDS), Sixth Year Production UH-6A

19. ABSTRACT *(Continue on reverse if necessary and identify by block number)*

Testing was conducted to establish a performance baseline of the 12th year production UH-60A helicopter. A total of 19.5 productive hours were flown at two different test sites between 15 August and 2 September 1988. The performance of the 12th year production UH-60A Black Hawk was similar to or slightly degraded from the sixth year production UH-60A. At the primary mission gross weight, the performance met the requirement of the Prime Item Development Specification (PIDS). The out-of-ground effect hover gross weight capability was 17,416 pounds for 95 percent intermediate (30 minute limit) rated power available at 4000 feet pressure altitude and 35°C temperature. The vertical rate of climb of the 12th year aircraft exceeded the PIDS requirement by 21 ft/min. The 12th year production aircraft has an increase in equivalent flat plate area over the sixth year production aircraft. At 4000 ft pressure altitude, 35°C, and maximum continuous power, the cruise airspeed is 139 knots true airspeed, equal to the PIDS requirement. (SDW)

20. DISTRIBUTION/AVAILABILITY OF ABSTRACT ☐ UNCLASSIFIED/UNLIMITED ☒ SAME AS RPT. ☐ DTIC USERS	**21. ABSTRACT SECURITY CLASSIFICATION** UNCLASSIFIED

22a. NAME OF RESPONSIBLE INDIVIDUAL	22b. TELEPHONE *(Include Area Code)*	22c. OFFICE SYMBOL
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INTRODUCTION

BACKGROUND

1. The current statement of baseline performance for UH-60A helicopters was based on data gathered during the airworthiness test of the Sixth Year Production UH-60A (ref 1, app A). With all the modifications to the Black Hawk airframe during the years since, an update of this information has been required. The U.S. Army Aviation Systems Command (AVSCOM) directed the U.S. Army Aviation Engineering Flight Activity to conduct a baseline performance evaluation of the 12th year production UH-60A helicopter (ref 2).

TEST OBJECTIVE

2. The objective of this test was to establish a performance baseline of the 12th year production UH-60A Black Hawk helicopter.

DESCRIPTION

3. The UH-60A Black Hawk helicopter is a twin-turbine, single main rotor helicopter capable of transporting internal and external cargo, 11 combat troops and weapons during day or night in visual or instrument meteorological conditions. Manufactured by Sikorsky Aircraft Division of United Technologies Corporation, the UH-60A has conventional wheel-type landing gear. The helicopter has a side-by-side pilot cockpit configuration with a conventional hydro-mechanical flight control system incorporating a dual stability augmentation system and limited attitude hold features. The pitch bias actuator in the later production aircraft has been replaced with a fixed link. The main and tail rotors are both four-bladed with the tail rotor mounted on the right side of the vertical tail pylon at a 20 degree upward cant. A movable horizontal stabilator located on the lower portion of the vertical tail pylon is programmed by the aircraft automatic flight control system. The helicopter is powered by two T700-GE-700 turboshaft engines having an uninstalled thermodynamic rating (30 minute) of 1584 shaft horsepower (shp) (power turbine speed of 20,900 revolutions per minute) each at sea level, standard day static conditions. Installed dual engine power is transmission limited to 2828 shp.

4. The UH-60A helicopter, USA S/N 88-26015, used for this evaluation was a 12th year production aircraft which incorporated the external stores support system fixed provisions and fairings, the reoriented production airspeed probes, the modified production stabilator schedule, the wire strike protection system and the production hover infrared suppressor subsystem (HIRSS). The HIRSS inner baffles were installed for all tests. The aft cabin vibration absorber has been removed for the later production helicopters including the test aircraft. Roll vibration absorbers were installed for this test in each of the main landing gear sponsons. The end fairing on each sponson was extended 4.75 inches to cover the vibration absorbers. For the purposes of the evaluation, an air data boom was installed on the lower right belly of the aircraft extending upward and forward beyond the nose of the helicopter. A movable ballast cart was installed in the aircraft to permit positive control of the longitudinal center of gravity in flight, compensating for the effects of fuel burned. A more detailed description of the

UH-60A is available in the Prime Item Development Specification (PIDS) (ref 3), the operator's manual (ref 4), and appendix B which includes a detailed description of the test configuration.

TEST SCOPE

5. This evaluation was conducted at Edwards AFB (elevation 2302 feet) and Bakersfield (elevation 507 feet), California, over the period from 15 August to 2 September 1988 and consisted of 20 flights totaling 19.5 productive flight hours. Limited level flight performance and a single hover performance flight were conducted. The test conditions are presented in table 1. The aircraft was operated within the limits of the operator's manual (ref 4) and the airworthiness release (ref 5) issued by AVSCOM. Testing was conducted in accordance with the test plan (ref 6). The test configuration was as specified by the PIDS (ref 3); AN/ALQ-144 infrared jammer mount removed, chaff dispenser and mounts removed, all windows and doors closed, the bleed air heater OFF and HIRSS inner baffle installed. Test results were compared with the results of previous UH-60A tests (refs 1 and 7).

TEST METHODOLOGY

6. Tethered hover test technique was used during the hover performance flight and the gathered data was used to calculate the vertical rate of climb. Flight test data was recorded by hand and by onboard magnetic tape. The Real Time Data Acquisition and Processing System was used for a portion of the flights conducted at Edwards AFB. Data was recorded from standard ship and sensitive calibrated instruments at the pilot's and flight test engineer's stations. The flight test engineer's station was installed as part of the instrumentation effort, a detailed listing of which is presented in appendix C. Specific test techniques and data analysis methods are presented in appendix D.

Table 1. Test Conditions[1]

Type	Gross Weight (lb)	Longitudinal Center of Gravity (FS)	Density Altitude (ft)	Referred Rotor Speed (rpm)	Trim Airspeed (KTAS)[2]
Airspeed Calibration	16710	347.6	8410	254	83 to 156 KIAS[3]
Hover Performance	13,210[4] to 20,640	358.0	3460	246 to 261	0
Level Flight Performance	15,310 to 20,830	347.6	3060 to 11,260	249 and 258	51 to 163

NOTES:

[1]Tests were conducted at an approximate mid-lateral center of gravity with the Automatic Flight Control System on in the normal utility configuration.
[2]KTAS: Knots true airspeed.
[3]KIAS: Knots indicated airspeed.
[4]Aircraft gross weight plus cable tension.

RESULTS AND DISCUSSION

GENERAL

7. Testing was conducted to obtain data for a baseline performance verification of the 12th year production UH-60A Black Hawk helicopter. The performance of the 12th year production UH-60A Black Hawk was similar to or slightly degraded from the sixth year production UH-60A. At the primary mission gross weight, the performance met the requirement of the Prime Item Development Specification (PIDS). At the performance guarantee conditions of 95 percent intermediate rated power (IRP) available (30 minute limit), 4000 feet pressure altitude (Hp) and 35°C, the out-of-ground effect (OGE) hover gross weight capability was 17,416 lb. The vertical rate of climb of the 12th year aircraft exceeded the PIDS requirement by 21 ft/min. The 12th year production aircraft has an increase in equivalent flat plate area over the sixth year production aircraft. At 4000 ft Hp, 35°C, and maximum continuous power, the cruise airspeed is 139 knots true airspeed (KTAS), equal to the PIDS requirement.

HOVER PERFORMANCE

8. Hover performance tests were conducted on the 12th year production aircraft at the conditions in table 1 using the tethered and free flight techniques described in appendix D. The 100 foot main wheel height OGE test was conducted at the 2302 foot test site in the normal utility configuration with the Hover Infrared Suppressor System installed. Tip mach number for this test varied from 0.78 to 0.82. The data from this test was compared with the sixth year production hover data presented in reference 1. The power required is slightly less than the sixth year data at the PIDS guarantee conditions of a coefficient of thrust equal to 0.007452. Results are presented in figure E-1.

9. At the performance guarantee conditions of 95 percent IRP, 4000 ft Hp, and 35°C, the OGE hover gross weight capability was 17,416 lb. IRP available was calculated from the General Electric engine deck (ref 8).

10. Nondimensional tail rotor power versus main rotor thrust in a hover is presented in figure E-2.

VERTICAL CLIMB PERFORMANCE

11. Vertical rate of climb was determined as described in appendix D from the OGE hover data at the performance guarantee conditions and a range of gross weights. Results are presented in figure E-3. A vertical rate of climb of 411 ft/min was found for the mission gross weight of 16,993.6 lb. This exceeds the PIDS requirement of 390 ft/min (ref 3) by 21 ft/min.

LEVEL FLIGHT PERFORMANCE

12. 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 12th year production

aircraft. The data were obtained in ball centered flight and corrected for instrumentation electrical load and estimated drag of external test instrumentation.

13. Nondimensional test results are presented in figures E-4 through E-8. Dimensional level flight test results are presented in figures E-9 through E-16. At the lower referred rotor speed (figs. E-9 through E-11) data shows more power required below approximately 120 KTAS when compared to the higher rotor speed. Previous data (refs 1 and 7) shows the same or less power required throughout the airspeed range. Based on a comparison with the previous test results, it was determined that the low rotor speed data was suspect. For this reason, fairings for the lower rotor speed data were based on the 258 rotor speed data. As a result, power required is similar for referred rotor speeds of 249.7 and 258.3 at coefficient of thrust up to 0.0080 and advance ratios up to 0.22. Beyond these values, power required increased with increased rotor speeds.

14. The primary mission of the 12th year Black Hawk is defined in the PIDS (ref 3) and stipulates the performance conditions of operating at 4000 ft Hp, 35°C, normal operating rotor speed (257.9 revolutions per minute), and a mission gross weight equal to 16,993.6 lb. At these conditions, using maximum continuous power, the PIDS states the cruise airspeed will be 139 KTAS. Figure E-17 represents the 12th year aircraft at the performance conditions. The data shows a cruise speed of 139.0 KTAS which meets the cruise airspeed requirements of the PIDS.

15. Tail rotor shaft horsepower (shp) was measured throughout the performance testing. Figures E-18 and E-19 represents tail rotor shp versus true airspeed for all speed power polars.

16. The increase in drag between the test aircraft and the sixth year aircraft as tested was 2.5 ft^2 equivalent flat plate area. The sixth year aircraft was tested with the brackets installed for the AN/ALQ-144(V) infrared countermeasures set and M-130 chaff dispenser (ref 1). The PIDS stipulated that the 12th year aircraft performance was without these brackets. Therefore, the total delta drag between the sixth year aircraft and the test aircraft is 4.0 ft^2. This difference was summarized as follows:

> 12th year production aircraft = difference as tested between sixth and 12th year aircraft (2.5 ft^2) + M-130 and ANALQ-144(V) brackets (1.5 ft^2).

The external components that produced an increase in equivalent flat plate area for the test aircraft were the Hover Infrared Suppression System (estimated 2 ft^2), Wire Strike Protection System (estimated 1 ft^2), and a combination of additional changes. The sponson vibration roll absorbers with fairings are expected to have a minimal effect on performance, however, this was not tested.

ENGINE PERFORMANCE

17. Figure E-20 presents a comparison of power transmitted through the main and tail rotor drive shafts and engine power output. The relationship between these powers equates to the system losses. These system losses ranged from 90 shp at 1200 engine shp

to 170 shp at 2800 engine shp. These results differ from previous testing (ref 9) for an unapparent reason.

AIRSPEED CALIBRATION

18. The standard ship's airspeed system on the 12th year production aircraft was calibrated in level flight. A calibrated T-34C pace aircraft and a calibrated trailing bomb were used to determine the position error. The position error of the ship's airspeed system is presented in figure E-21.

CONCLUSIONS

GENERAL

19. The performance of the 12th year production UH-60A Black Hawk was similar to or slightly degraded from the sixth year production UH-60A. At the primary mission gross weight, the performance met the requirement of the Prime Item Development Specification (PIDS) (para 7).

SPECIFIC

20. The out-of-ground effect (OGE) hover power required is slightly less than the previous sixth year OGE data at the PIDS conditions of coefficient of thrust equal to 0.007452 (para 8).

21. At the performance guarantee conditions, the OGE hover gross weight capability was 17,416 lb (para 9).

22. The 12th year aircraft demonstrated an increase in equivalent flat plate area of 4.0 ft^2 over the sixth year aircraft (para 14).

SPECIFICATION COMPLIANCE

23. Paragraph 3.2.1.1.1.1a. A vertical rate of climb of 411 ft/min was found for the mission gross weight and this exceeds the PIDS requirement of 390 ft/min by 21 ft/min (para 10).

24. Paragraph 3.2.1.1.1.1b. At the performance conditions and using maximum continuous power, a cruise speed of 139.0 knots true airspeed exists which meets the PIDS requirement (para 13).

RECOMMENDATION

25. None.

APPENDIX A. REFERENCES

1. Final Report, AEFA Project No. 83-24, *Airworthiness and Flight Characteristics Test of a Sixth Year Production UH-60A*, June 1985.

2. Letter, Headquarters, AVSCOM, 6 January 1988, AMSAV-8, subject: Baseline Performance Verification of the UH-60A Black Hawk Helicopter. (Test Request)

3. Prime Item Development Specification (PIDS), Sikorsky Aircraft Division, DARCOM-CP-2222-S1000H, *UH-60A Black Hawk Aircraft*, 11 December 1987.

4. Technical Manual, Headquarters Department of the Army, TM 55-1520-237-10, *Operator's Manual, UH-60A and EH-60A Helicopter*, 8 January 1988, with change 1 dated 29 March 1988.

5. Letter, Headquarters AVSCOM, AMSAV-E, 21 July 1988, subject: Airworthiness Release for UH-60A Black Hawk Helicopter S/N 88-26015 to Conduct an Airworthiness and Flight Characteristics (A&FC) Test of the Twelfth Year Production UH-60A, AEFA Project No. 87-32.

6. Test Plan, AEFA Project No. 87-32, *Baseline Performance Verification of the 12th Year Production UH-60A Black Hawk Helicopter*, February 1988.

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

8. General Electric Engine Deck Number 84127A, dated 6 June 1984.

9. Final Report, AEFA Project No. 77-23, *Production Validation Test Government (PVT-G), Performance Guarantees UH-60A Black Hawk Helicopter*, October 1979.

10. Final Report, AEFA Project No. 87-07, *Airworthiness and Flight Characteristics Evaluation of the EH-60A (Quick Fix) Helicopter*, October 1988.

11. Final Report, AEFA Project No. 74-06-1, *Government Competitive Test Utility Tactical Transport Aircraft System (UTTAS) Sikorsky YUH-60A Helicopter*, November 1976.

APPENDIX B. 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 (figs. B-1 through B-4). A complete description of the aircraft is contained in the operator's manual (ref 4, app A).

2. The helicopter used for this evaluation (USA S/N 88-26015) is a 12th year production aircraft. The following photographs illustrate the configuration differences between the sixth year production helicopter (ref 1) and the 12th year production helicopter used in this evaluation.

EXTERNAL STORES SUPPORT SYSTEM FIXED PROVISION FAIRINGS

3. The 12th year production aircraft is equipped with provisions for incorporating the External Stores Support System (ESSS). With the system removed, aerodynamic fairings are installed. 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 sixth year production aircraft (ref 1) was configured with the ESSS fixed provisions.

HOVER INFRARED SUPPRESSOR SUBSYSTEM

4. The Hover Infrared Suppressor Subsystem (HIRSS) (figs. B-5 and B-6) has no moving parts. It reduces the helicopter's infrared (IR) signature by mixing ram air with the engine exhaust gases, and by blocking line-of-sight view of hot metal parts. The IR suppressor channels exhaust gases through a sheet metal core mounted within a fiberglass honeycomb sandwich-constructed nacelle. The suppressor core is constructed of short segments; each successive segment, in the direction of gas flow, has a larger cross-sectional area than the previous one. The inside surface of each segment is coated with low-reflectance material. Cooling air, entering the ram inlet scoop, is ducted around the suppressor core and passes through the gaps between overlapping core segments, providing film-cooling of the core surface. The engine exhaust plume is cooled internally by mixing the core film-cooling air, and externally by crossflow mixing with ambient freestream at the suppresser exit. The core turns outboard and downward to prevent line-of-sight seeking of the hot engine turbine and rear frame and to direct the engine exhaust into the cooling air of the free-stream and rotor downwash. The HIRSS was not part of the sixth year production aircraft.

WIRE STRIKE PROTECTION SYSTEM

5. The Wire Strike Protection System (WSPS) is a positive system used to cut, break or deflect wires that may strike the aircraft. Wire strike protection (figs. B-7 through B-9)

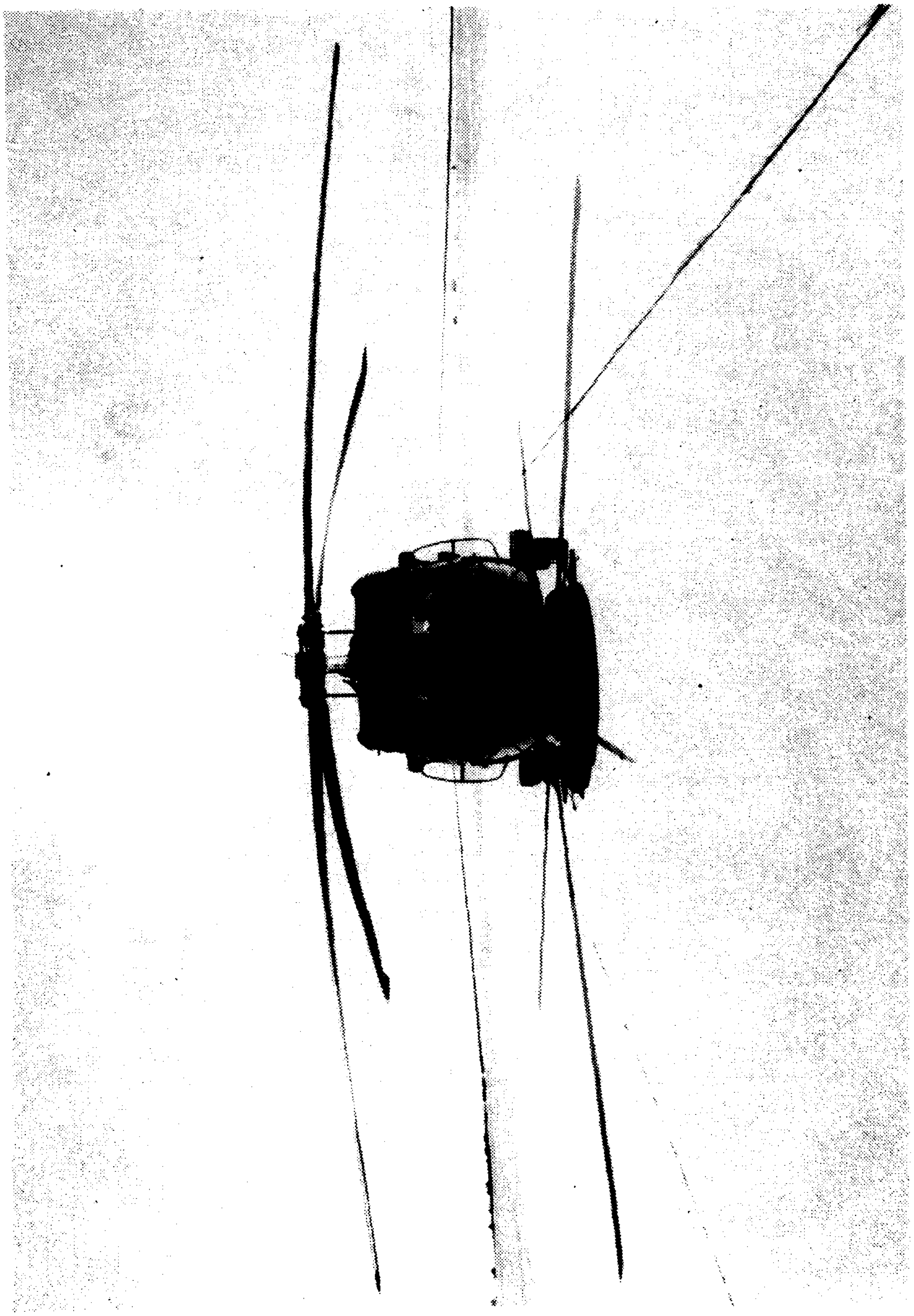


Figure B-1. Test Aircraft, Front View

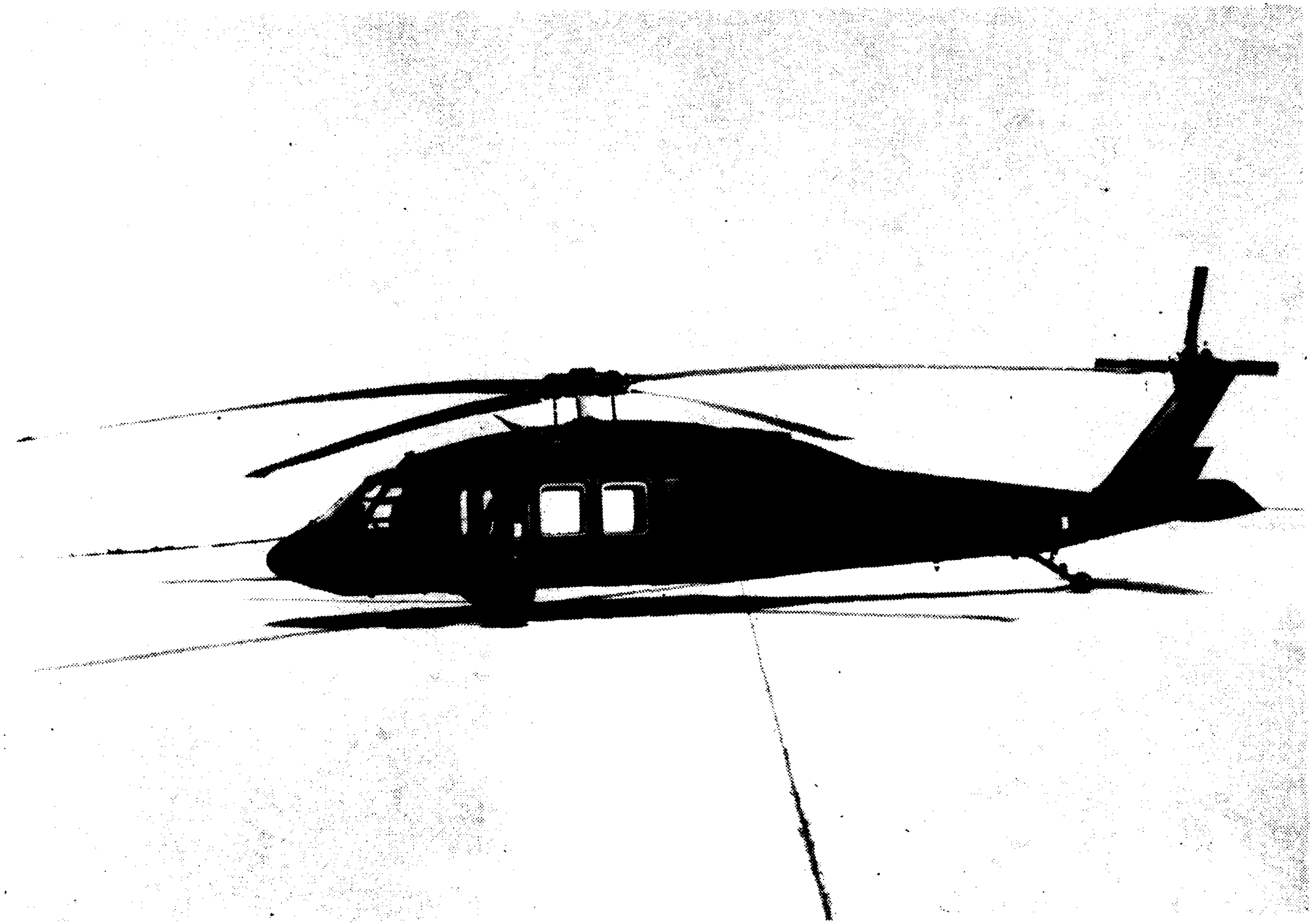


Figure B-2. Test Aircraft, Left Side View

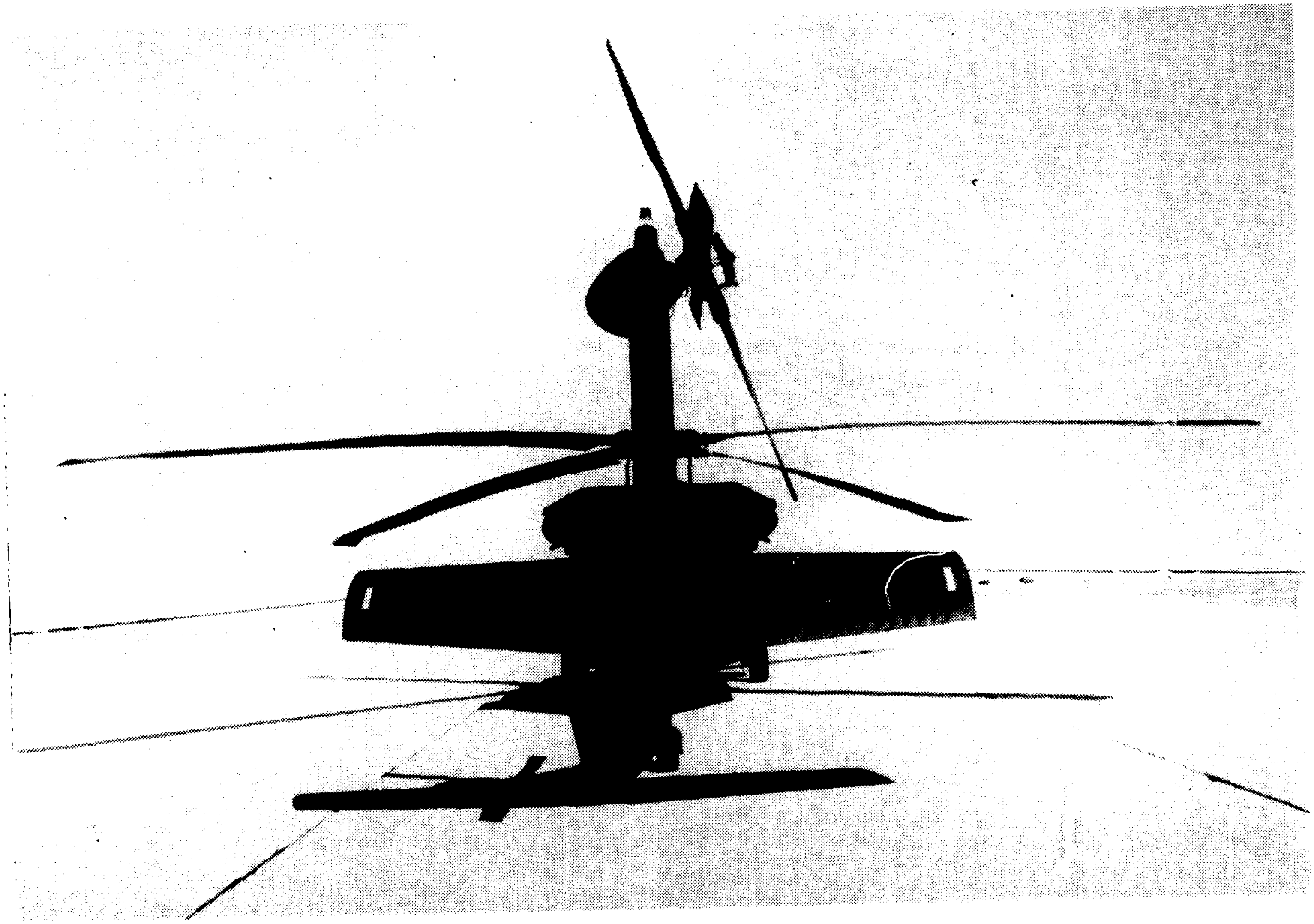


Figure B-3. Test Aircraft, Rear View

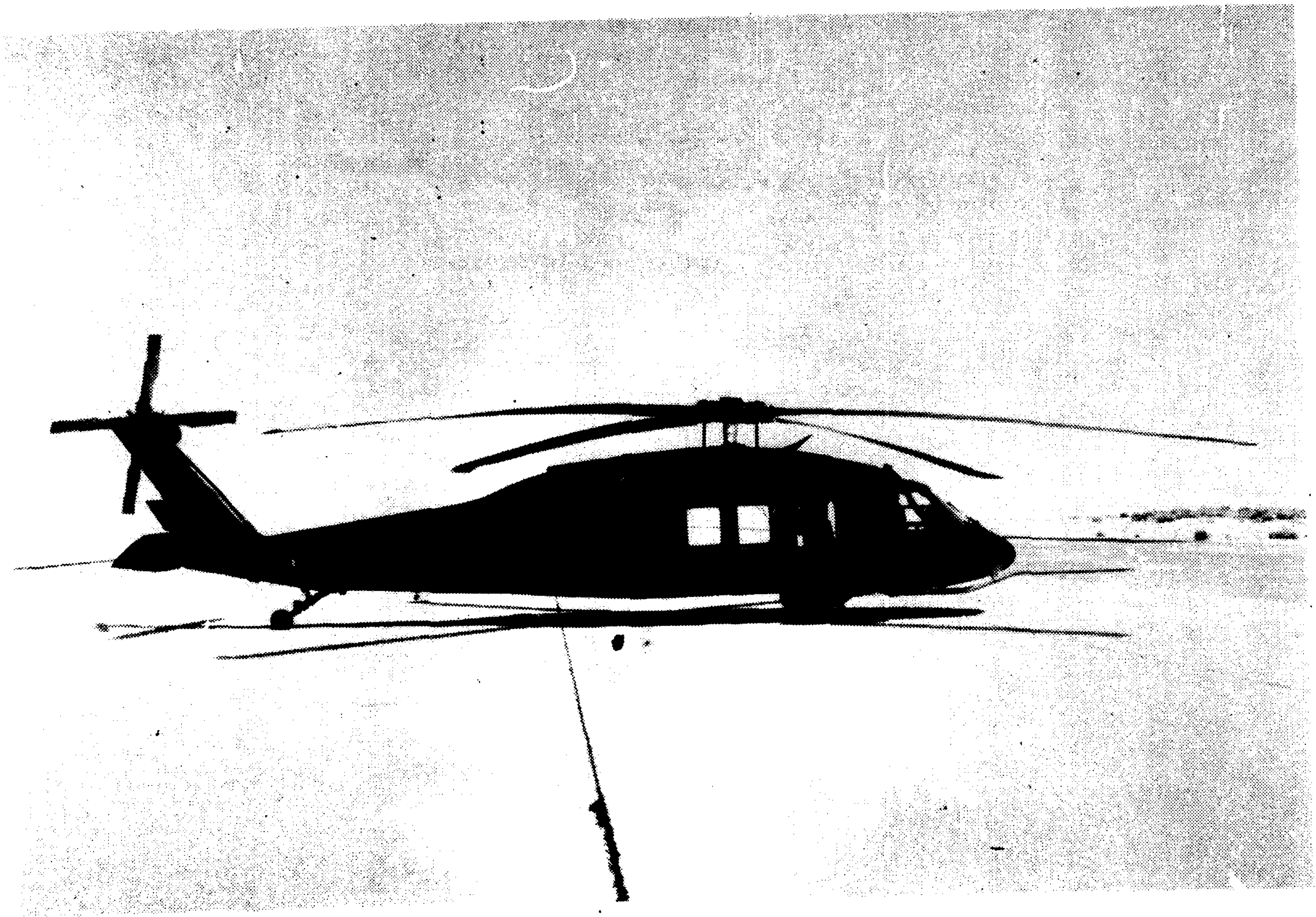


Figure B-4. Test Aircraft, Right Side View

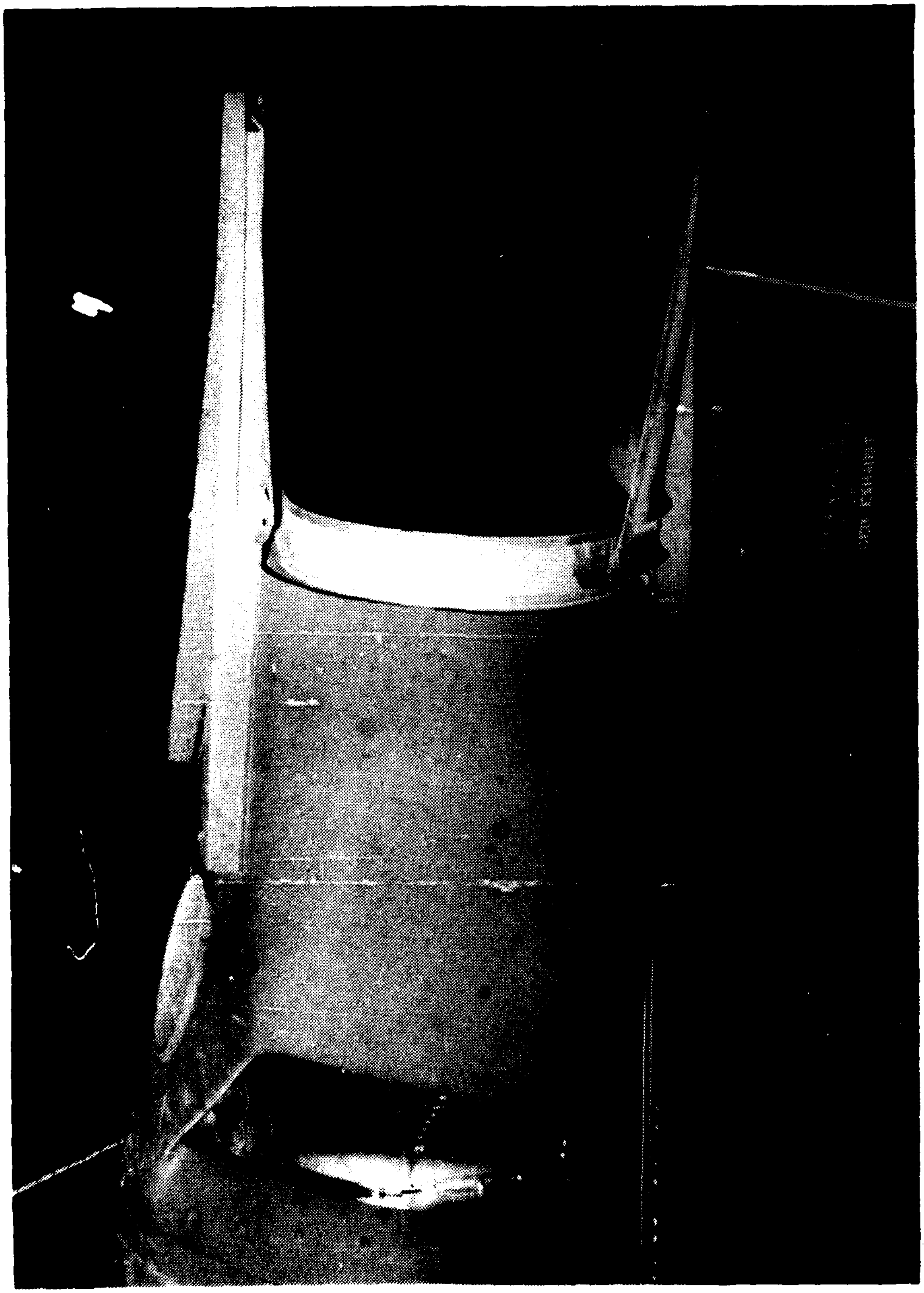


Figure B-5. Hover Infrared Suppressor Subsystem (Side View)

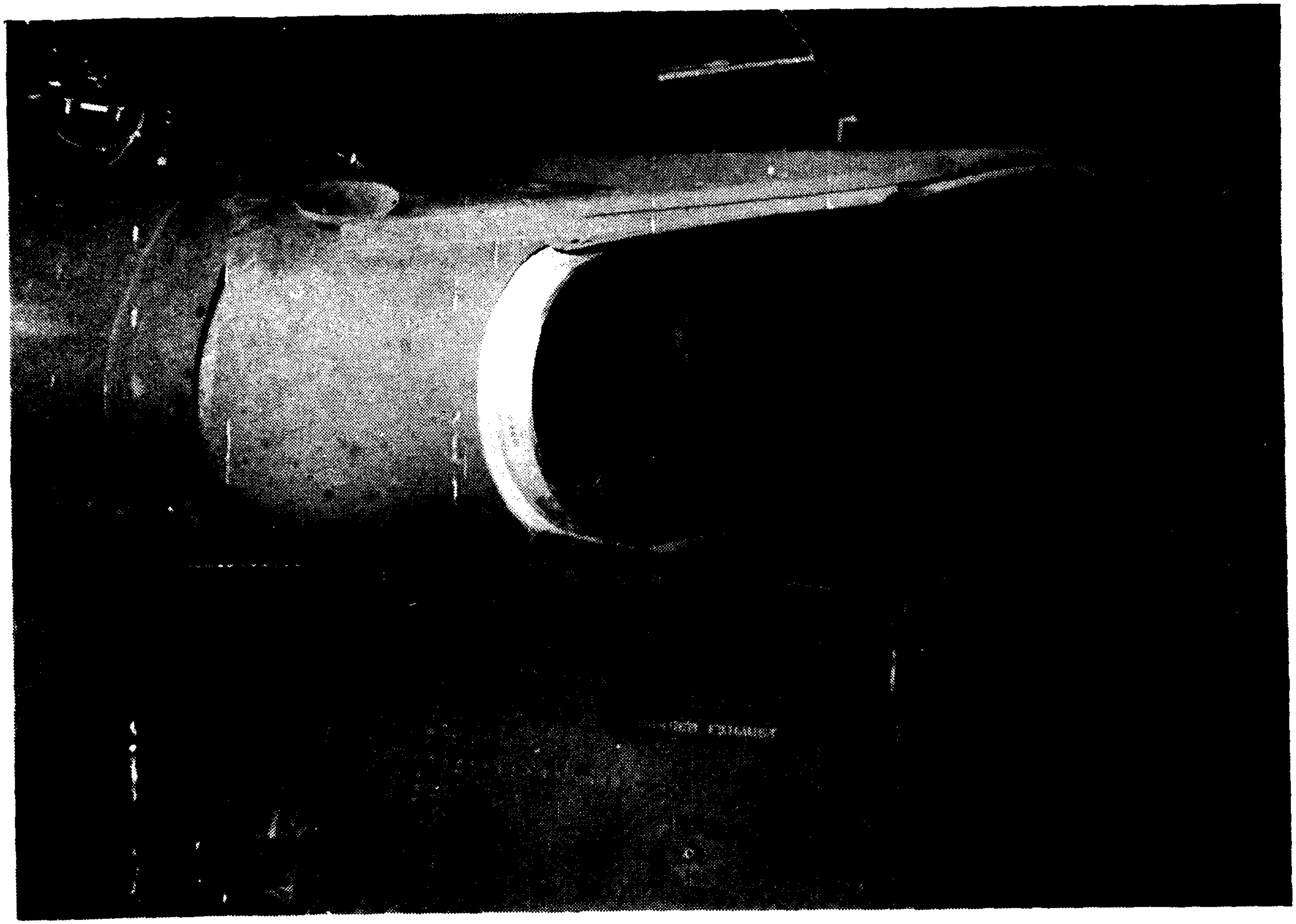


Figure B-6. Hover Infrared Suppressor Subsystem (Rear Exhaust View)

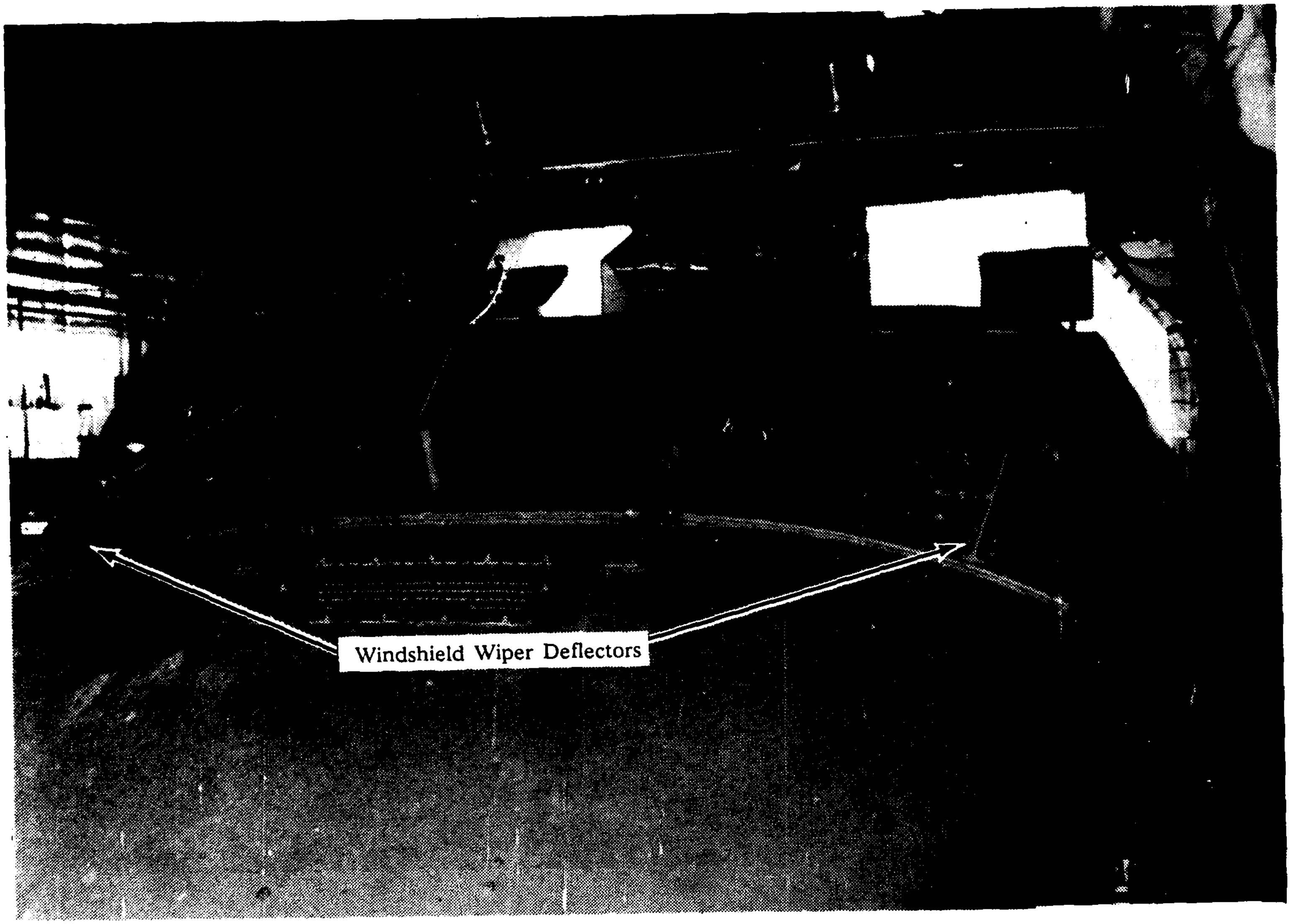


Figure B-7. Wire Strike Protection System – Windshield Wiper Deflectors

Figure B-8. Wire Strike Protection System - Pylon Cutter

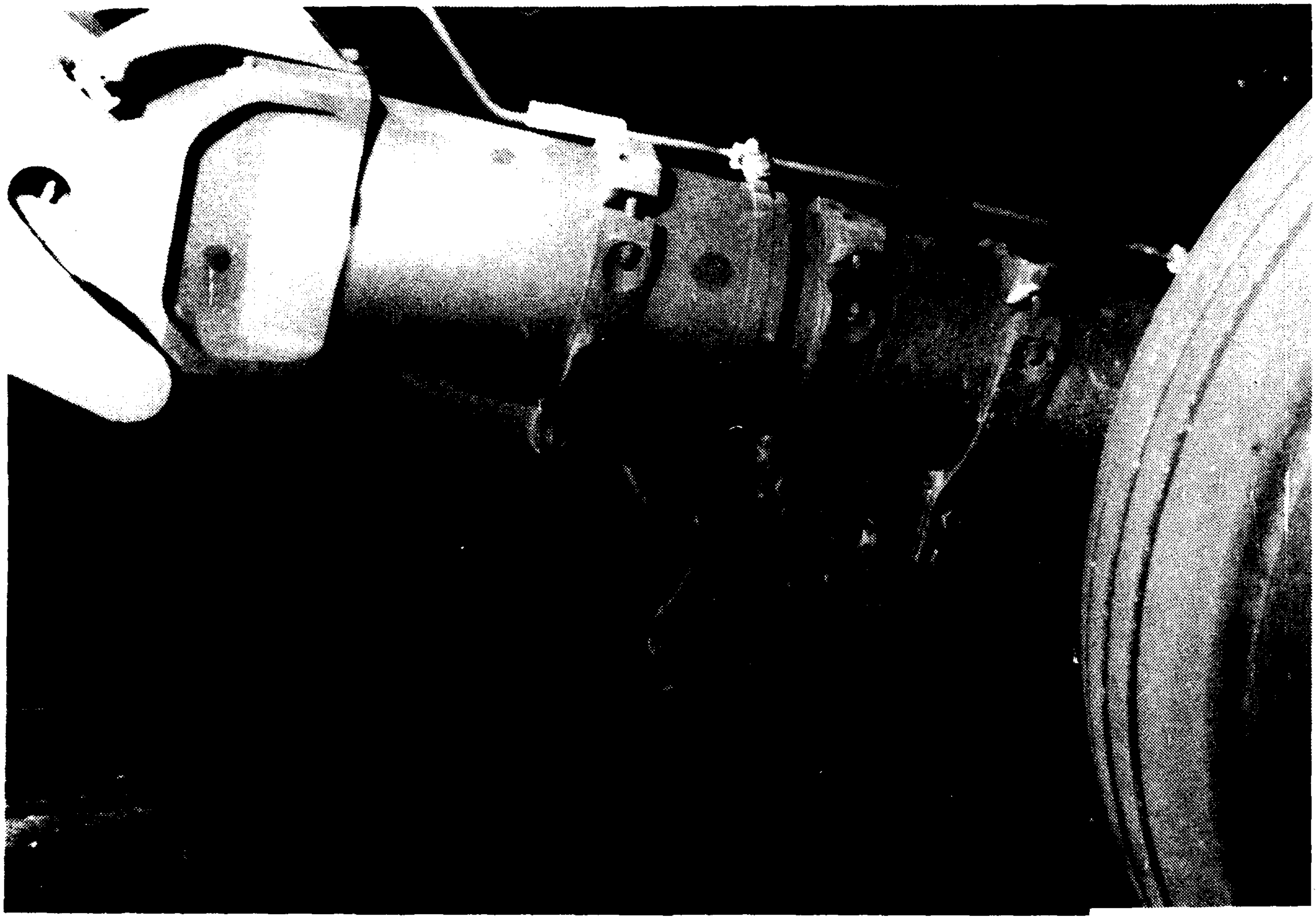


Figure B-9. Wire Strike Protection System – Main Landing Gear Deflector/Cutter exists for the frontal area between the tires and fuselage, between the fuselage and main rotor, and windshield wipers in level flight. The WSPS was not part of the sixth year production aircraft.

MISCELLANEOUS

6. The sponson vibration roll absorbers were installed on the test aircraft. Figures B-10 and B-11 illustrate the external difference between the test aircraft and sixth year production aircraft.

7. Other drag producing components are the outside air temperature probes. The 12th year production aircraft utilizes two temperature probes, one on either side of the aircraft nose, on the upper portion of each windshield. The sixth year production aircraft only utilized one temperature probe located between the windshields.

8. The brackets for the AN/ANQ-144(V) IR countermeasures set and the M-130 chaff/flare dispenser were removed from the 12th year aircraft for testing. The brackets remained for testing during the evaluation of the sixth year production aircraft.

ENGINES

9. The primary power plants for the UH-60A helicopter are General Electric T700-GE-700 front drive turboshaft engines, rated at 1584 shaft horsepower (shp) at a power turbine speed of 20,900 revolutions per minute (rpm) (sea level, standard day installed). The engine has four modules: cold section, hot section, power turbine section, and accessory section. Design features include an axialcentrifugal flow compressor, a through-flow combustor, a two-stage uncooled power turbine, and self contained lubrication and electrical system. Pertinent engine data are shown below:

Model	T700-GE-700
Type	Turboshaft
Rated power	1584 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

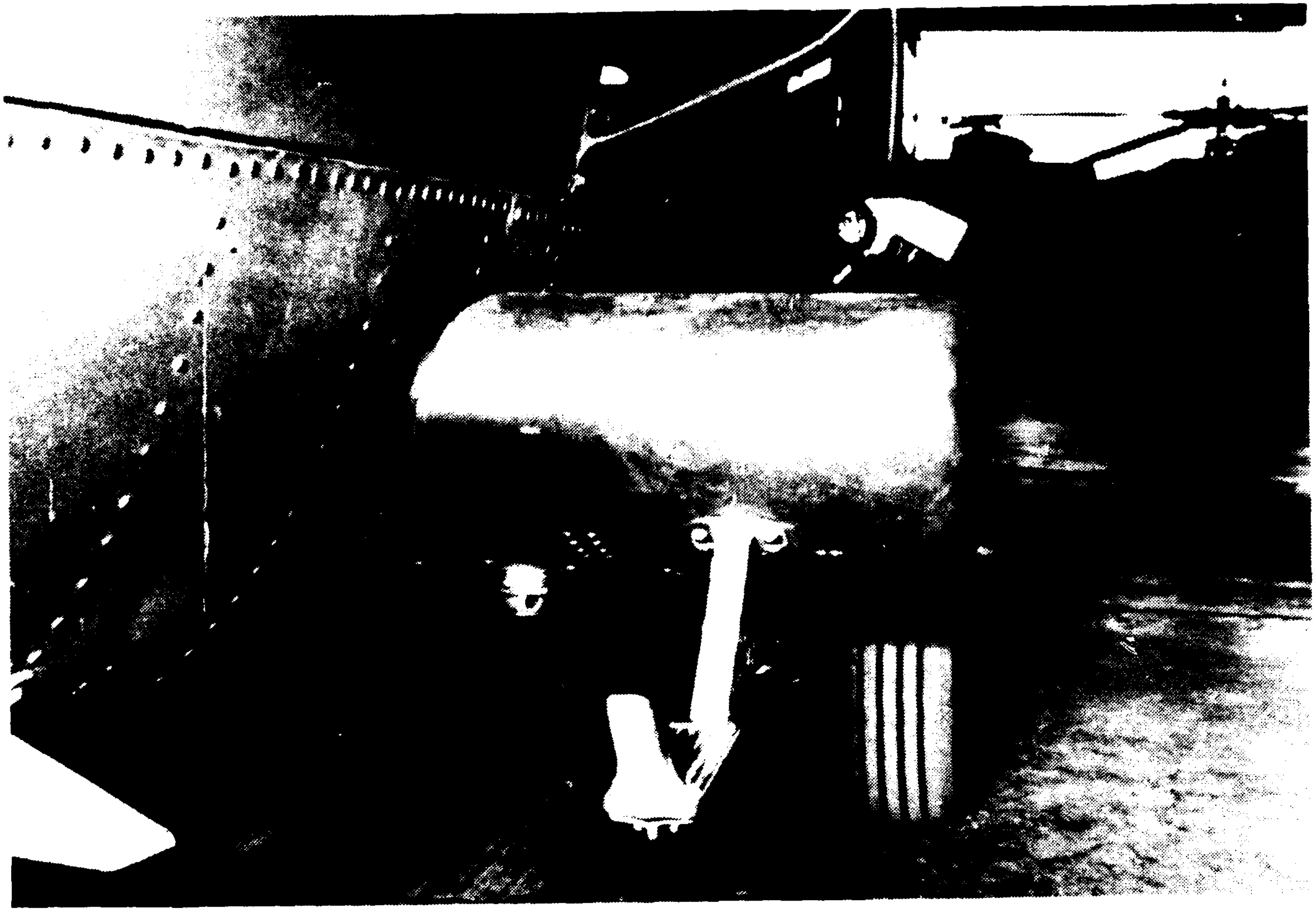


Figure B-10. Test Aircraft - Frontal View of the Sponson Vibration Roll Absorber Fairing

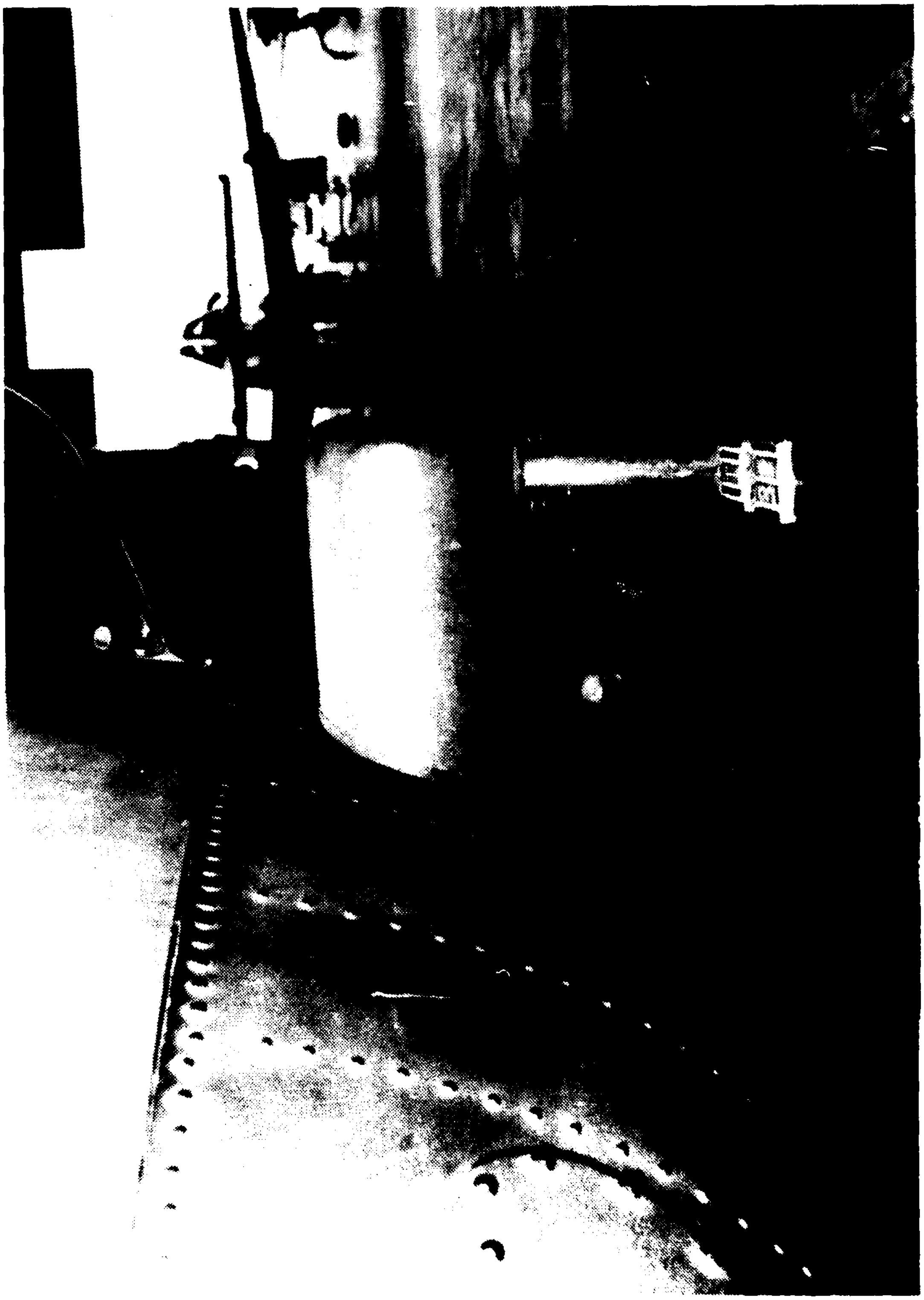


Figure B-11. Sixth Year Aircraft – Frontal View of the Sponson Fairing

BASIC AIRCRAFT INFORMATION

10. General data of the 12th year production UH-60A helicopter are as follows:

Gross Weight

Empty weight	Approximately 11,154 pounds
Primary Mission gross weight	16,993.6 pounds
Fuel capacity (measured)	364 gallons

Main Rotor

Number of blades	4
Diameter	53 ft, 8 in.
Blade chord	1.73 ft (root) 1.75 ft (droop snoop)
Blade twist (equivalent linear)	-18 deg
Blade tip sweep	20 deg aft
Blade area (one blade)	46.7 square feet
Airfoil	
Section (root to tip designation)	SC1095/SC1095R8
thickness (percent chord)	9.5%
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 square feet
Airfoil	
section (root to tip designation)	SC1095/SC1095R8
thickness (percent chord)	9.5%
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	$\left(\frac{228 + 62}{62}\right)$
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 U.S. Army Aviation Engineering Flight Activity. A test boom was installed on the right side belly of the aircraft, extending 98 inches forward of the nose. The boom incorporated an angle-of-attack sensor, angle-of-sideslip sensor, and a swiveling pitot-static tube. Additionally, the main rotor and tail rotor were configured with slipring assemblies for instrumentation purposes (figs. C-1 and C-2).

2. The data acquisition system utilized pulse code modulation (PCM). The mainframe sample rate was 454 samples/second while subframe sample rates varied from 28 to 114 samples/second. Data was obtained from telemetry, calibrated cockpit instruments or recorded on magnetic tape.

3. Data displayed on board the aircraft include the following.

Pilot Station

Airspeed (boom)
Airspeed*
Altitude (boom)
Altitude*
Altitude (radar)*
Rate of climb*
Rotor speed (digital)
Engine torque* **
Turbine gas temperature * **
Power turbine speed (Np)* **
Gas producer speed (Ng)* **
Horizontal stabilator position*
Center of gravity (cg) lateral acceleration (sensitive)
Angle of sideslip
Tether cable angles
    Longitudinal
    Lateral
Event switch

Copilot Panel

Airspeed*
Altitude*
Rotor speed*
Engine torque* **
Ballast cart control
Ballast cart position

* Non-calibrated standard ship instrument
**Both engines

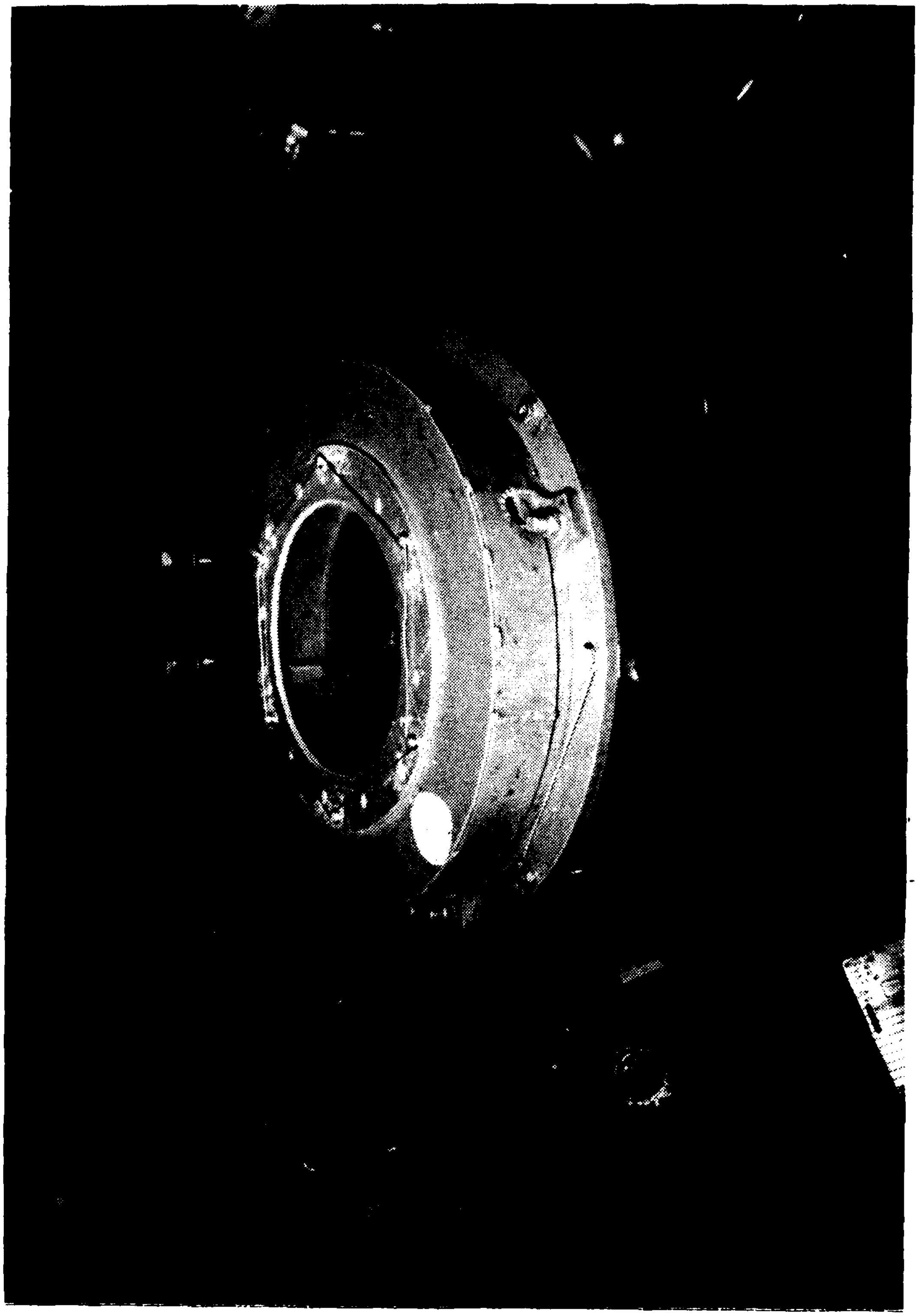


Figure C-1. Main Rotor Slipring Assembly

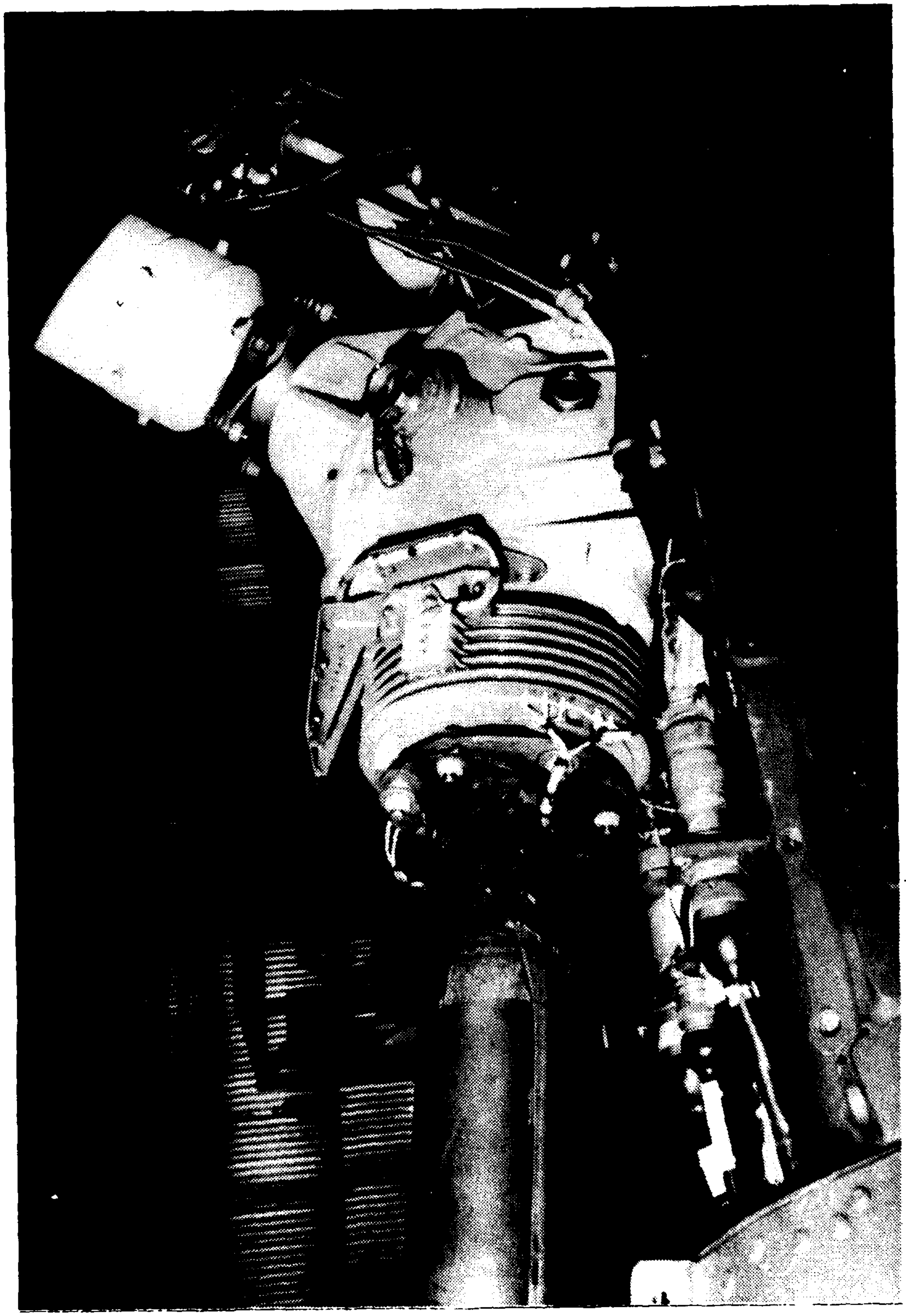


Figure C-2. Tail Rotor Slipring Assembly

Cable tension
Fuel remaining* **

4. The flight test engineers panel was installed as part of the instrumentation effort (fig. C-3).

Engineer Panel

Engine fuel flow**
Engine fuel used**
APU fuel used
Total air temperature
Rotor speed (digital)
Instrumentation controls
Single channel decommutator
Time code display
Run number
Event switch

5. Parameters recorded onboard the aircraft and via telemetry include the following.

Digital (PCM) Parameters

Airspeed (boom)
Altitude (boom)
Airspeed (ship's)
Altitude (ship's)
Total air temperature
Main rotor speed
Gas generator speed **
Power turbine speed **
Engine fuel flow **
Engine fuel used**
Engine fuel temperature **
Engine output shaft torque**
Engine turbine gas temperature**
APU fuel used
Main rotor shaft torque
Center of gravity acceleration
    Normal
    Lateral
    Vertical
CG lateral acceleration (sensitive)
Tether cable tension
Tether cable angle

*Non-calibrated standard ship instrument
**Both Engines

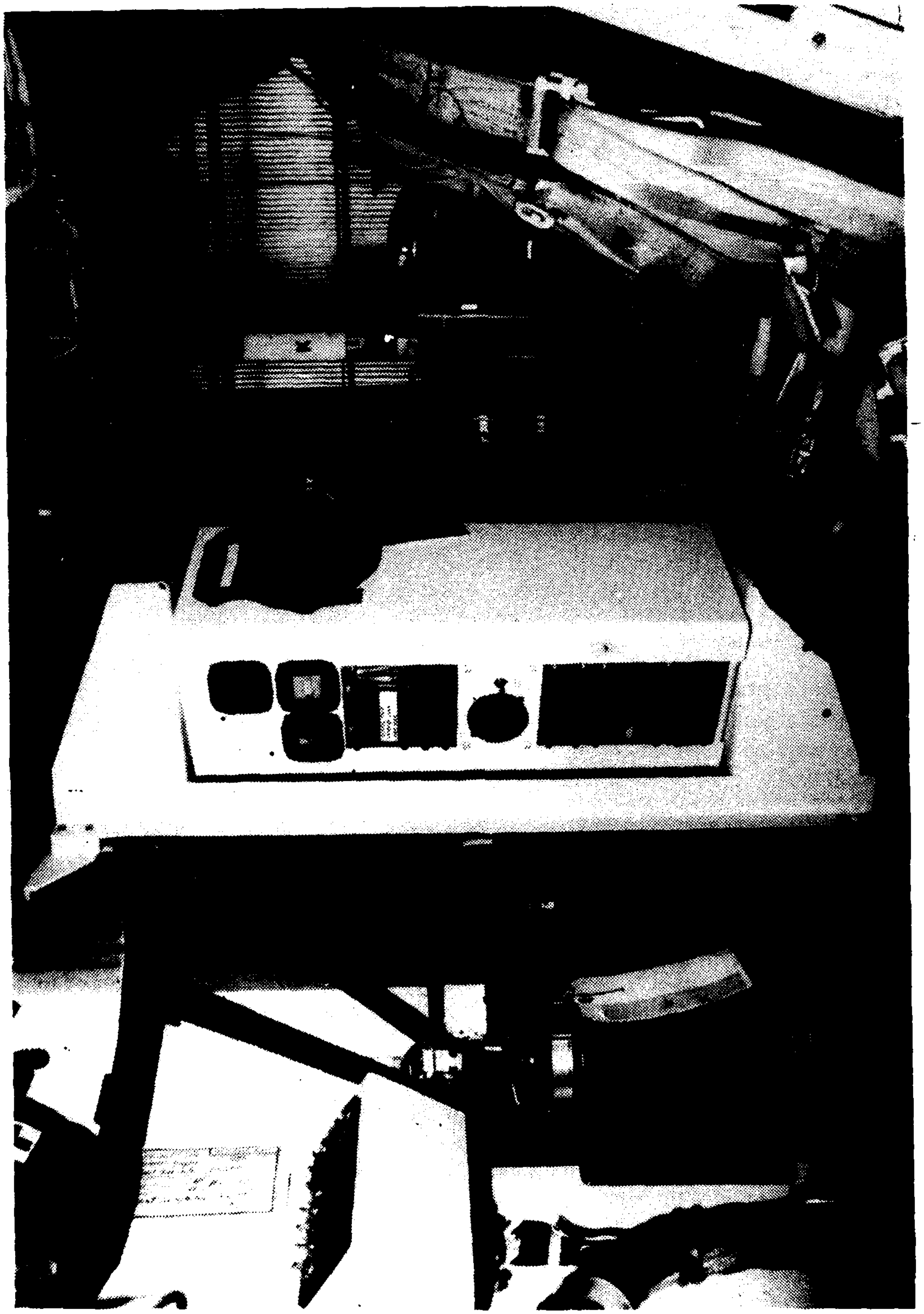


Figure C-3. Instrumented Engineer's Panel

Longitudinal
Lateral
Stabilator position
Movable ballast location
Attitude
Pitch
Roll
Tail rotor shaft torque
Angle of sideslip
Angle of attack
Time of day
Run number
Pilot event
Engineer event

Airspeed Calibration

6. The standard ship's airspeed system and test boom airspeed system were calibrated in level flight. A calibrated T-34C pace aircraft was used to determine the position error. The position error of the boom airspeed system is presented in figure C-4. The fairing used was from AEFA Project No. 87-07 (ref 10, app A) since the same boom system was used and an added low speed fairing (dashed line) was available.

Engine Calibration

7. Two T700-GE-700 engines and their accompanying Electronic Control Units were calibrated by Corpus Christi Army Depot located in Corpus Christi, Texas. The engine calibration data are presented in figures C-5 through C-10.

SPECIAL EQUIPMENT

Weather Station

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

Load Cell

9. 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 C-4
BOOM AIRSPEED CALIBRATION IN LEVEL FLIGHT
UH-60A USA S/N 88-26015

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
16710	347.6(FWD)	0.0	8410	23.5	258	T-34C PACE

NOTES:
1. NORMAL UTILITY CONFIGURATION
2. BALL CENTERED FLIGHT
3. FAIRING OBTAINED FROM AEFA PROJECT 87-07
4. DASHED LINE DENOTES LOW SPEED FAIRING

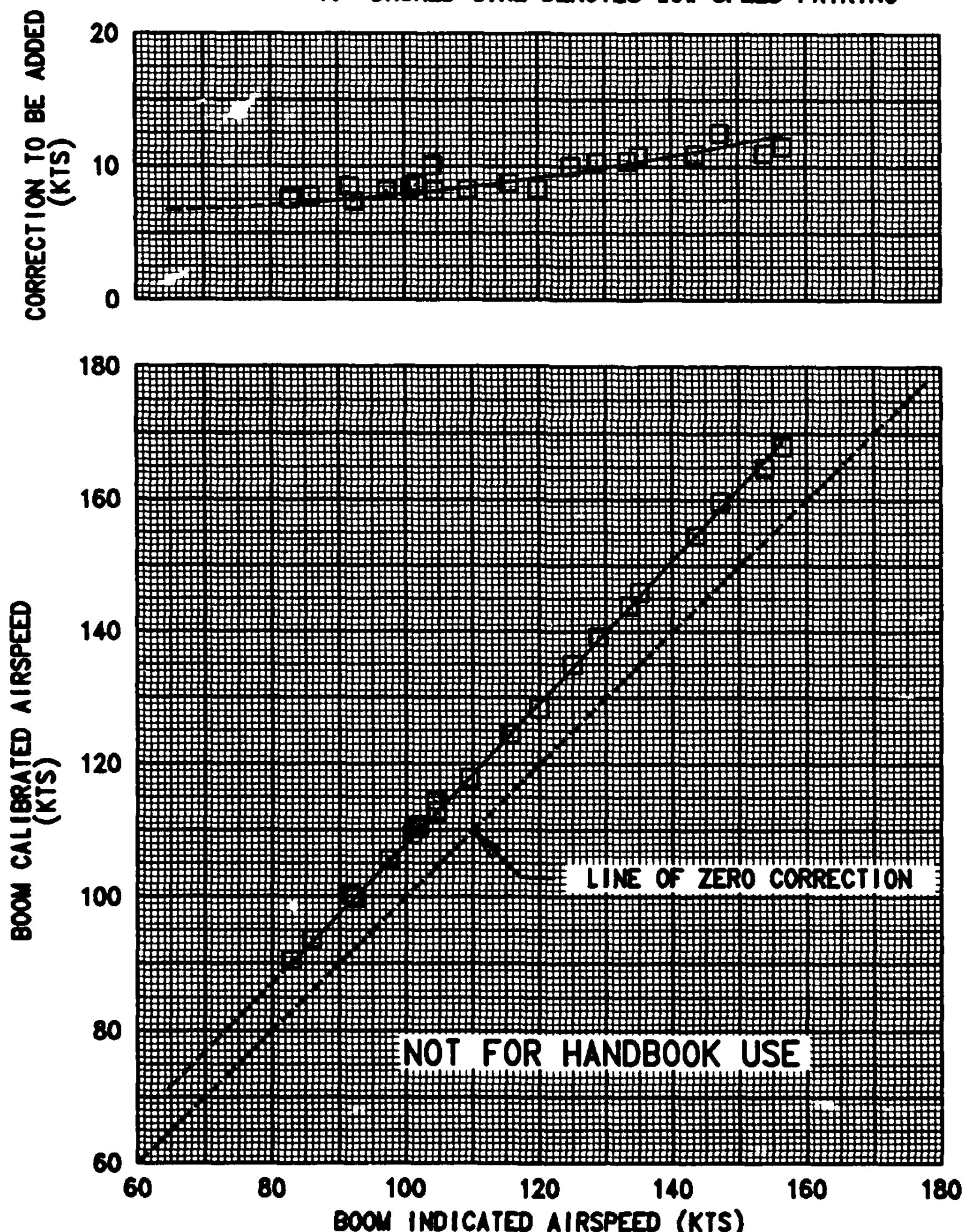


FIGURE C-5

ENGINE CALIBRATION

ENGINE S/N 307601

NOTES: 1. POWER TURBINE SPEED = 20900 RPM
2. OPEN SYMBOLS DENOTE INCREASING TORQUE
3. CROSSED SYMBOLS DENOTE DECREASING TORQUE

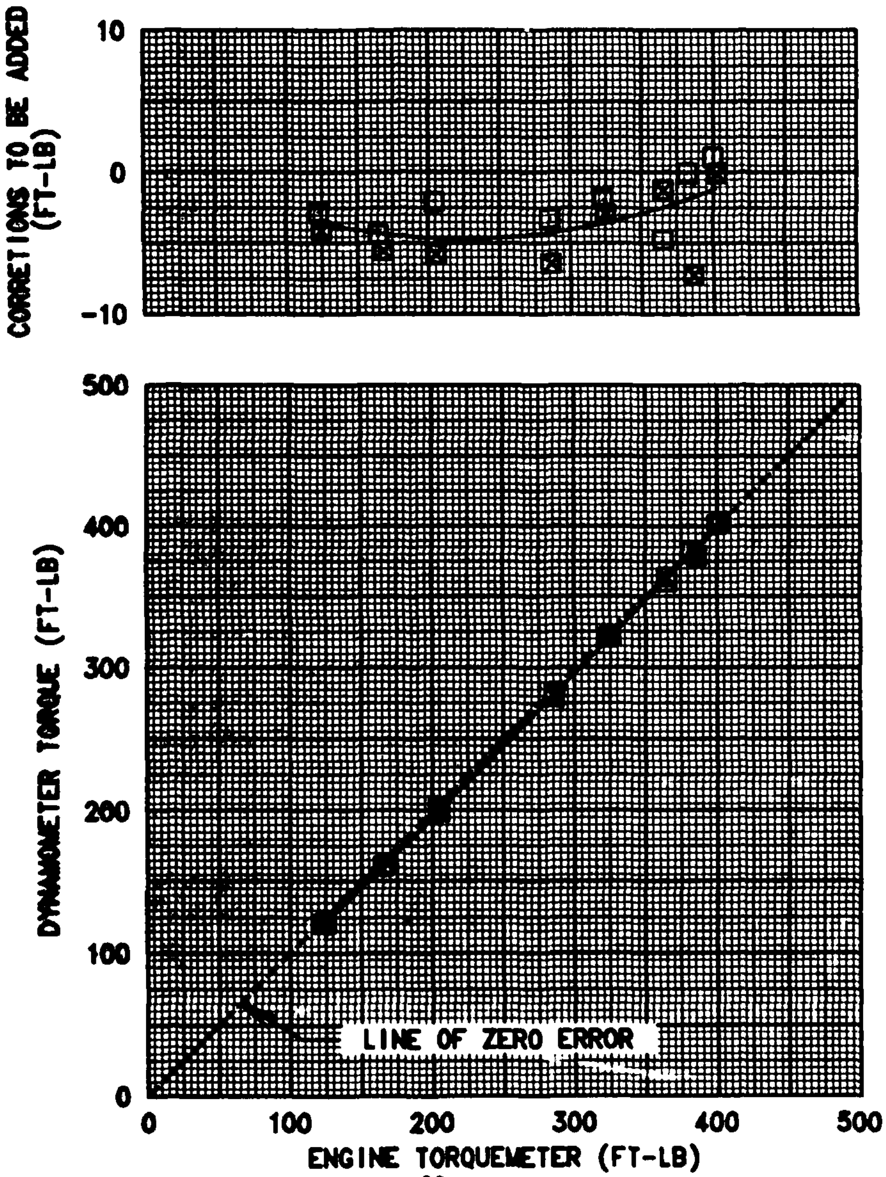


FIGURE C-6

ENGINE CALIBRATION

ENGINE S/N 307601

NOTES: 1. POWER TURBINE SPEED = 20380 RPM
2. OPEN SYMBOLS DENOTE INCREASING TORQUE
3. CROSSED SYMBOLS DENOTE DECREASING TORQUE

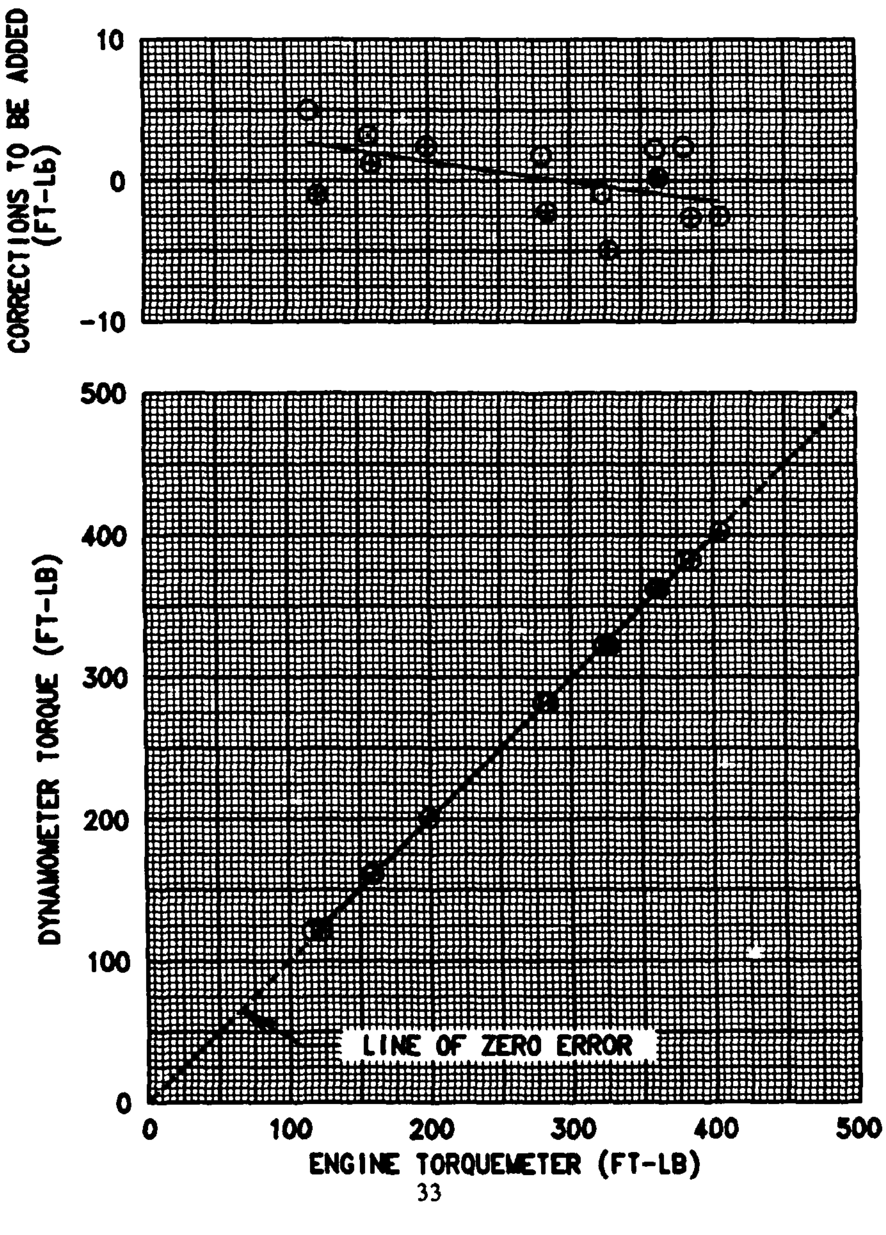


FIGURE C-7

ENGINE CALIBRATION

ENGINE S/N 307601

NOTES:
1. POWER TURBINE SPEED = 19870 RPM
2. TRIANGULAR SYMBOLS DENOTE INCREASING TORQUE
3. DIAMOND SYMBOLS DENOTE DECREASING TORQUE

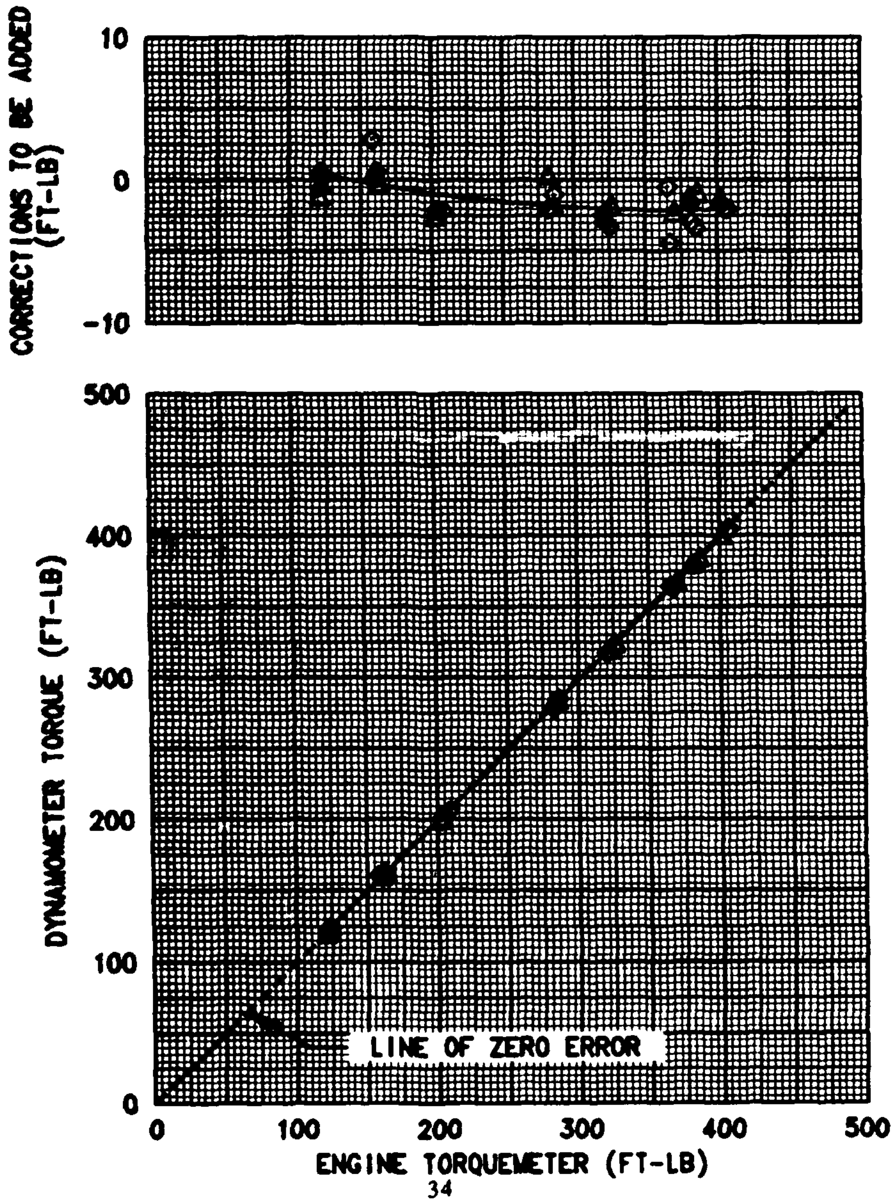


FIGURE C-8

ENGINE CALIBRATION

ENGINE S/N 307609

NOTES: 1. POWER TURBINE SPEED = 20900 RPM
2. OPEN SYMBOLS DENOTE INCREASING TORQUES
3. CROSSED SYMBOLS DENOTE DECREASING TORQUES

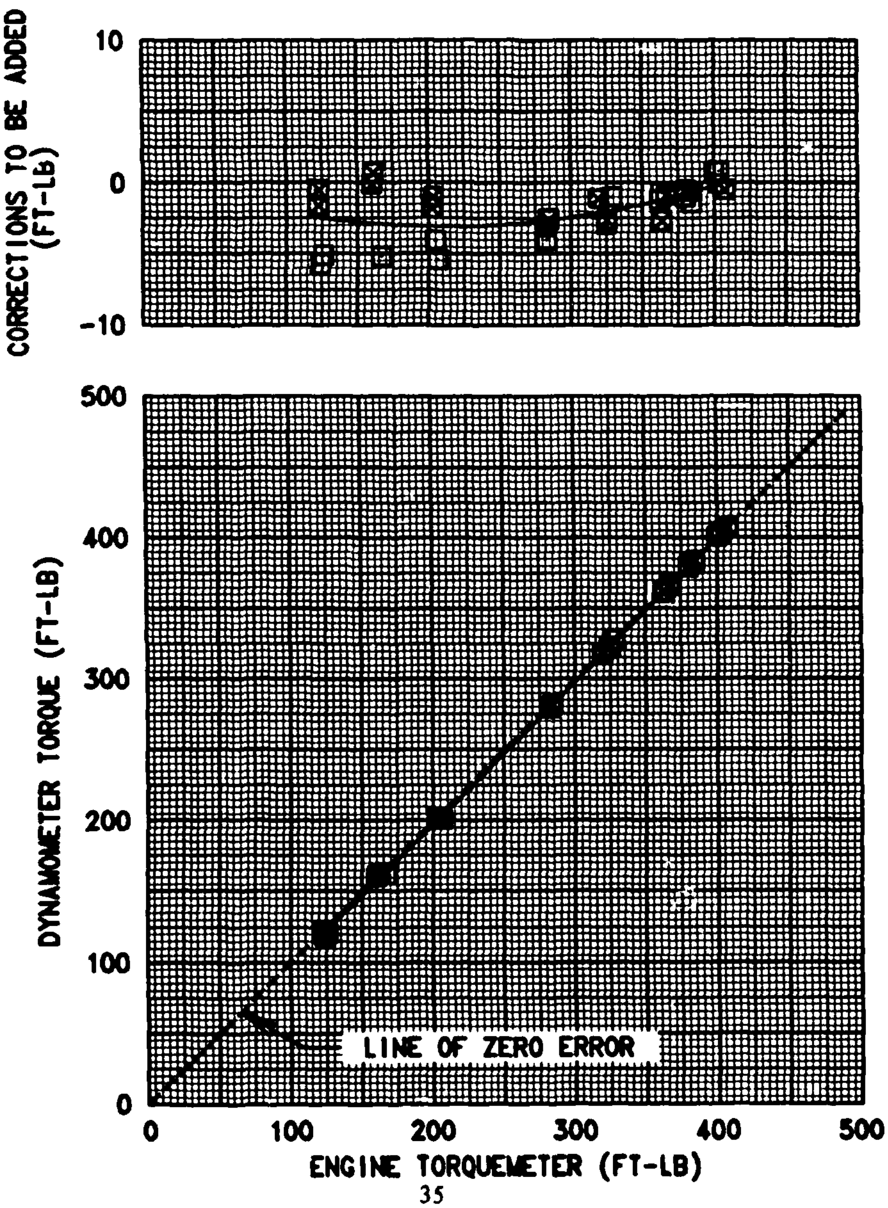


FIGURE C-9

ENGINE CALIBRATION

ENGINE S/N 307609

NOTES: 1. POWER TURBINE SPEED = 20380 RPM
2. OPEN SYMBOLS DENOTE INCREASING TORQUES
3. CROSSED SYMBOLS DENOTE DECREASING TORQUES

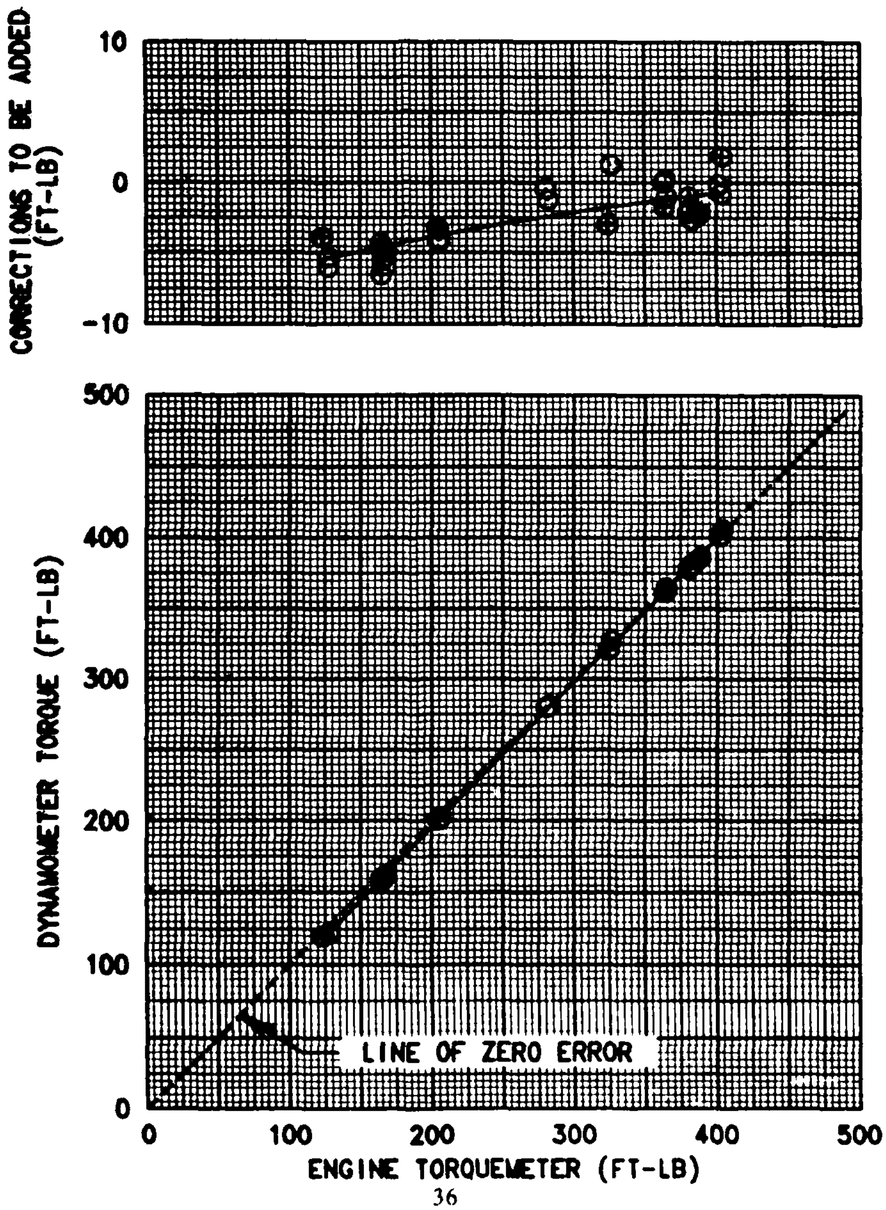
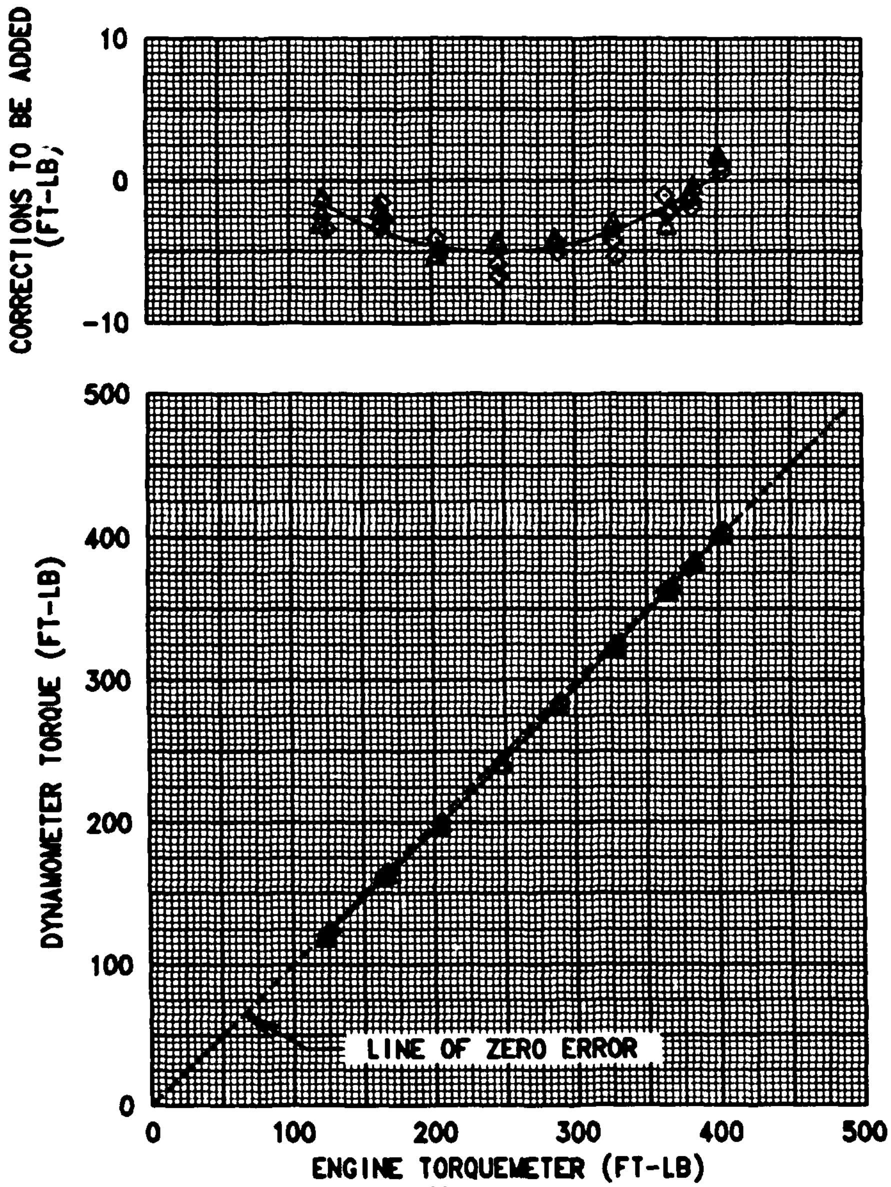


FIGURE C-10

ENGINE CALIBRATION

ENGINE S/N 307609

NOTES: 1. POWER TURBINE SPEED = 19870 RPM
2. TRIANGULAR SYMBOLS DENOTE INCREASING TORQUES
3. DIAMOND SYMBOLS DENOTE DECREASING TORQUES



APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

AIRCRAFT RIGGING

1. An engineering rigging check of the flight controls was performed to insure compliance with established limits. The stabilator was also checked and the results closely duplicated the modified production schedule.

AIRCRAFT WEIGHT AND BALANCE

2. The aircraft was weighed prior to the start of the test in the instrumented configuration with all fuel drained and full oil. The initial weight of the 12th year production aircraft was 12,171 pounds with the longitudinal center of gravity (cg) located at fuselage station (FS) 356.77 with the cg of the empty ballast cart located at FS 301.25. An external fuel quantity gage was also calibrated. The measured fuel capacity using the gravity fueling method was 364 gallons. The fuel weight for each test flight was determined prior to engine start and after engine shutdown by using the external 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 the results were compared with the sight gage readings. Aircraft cg was controlled by a movable ballast system which was manually positioned to maintain a constant cg while 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 72.0 inches.

PERFORMANCE

General

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

a. Coefficient of Power (C_P):

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

b. Coefficient of Power Tail Rotor ($C_{P_{TR}}$):

$$C_{P_{TR}} = \frac{SHP_{TR}(550)}{\rho A_{TR}(\Omega R)^3_{TR}} \quad (2)$$

c. Coefficient of Thrust (C_T):

$$C_T = \frac{GW + CABLE\ TENSION}{\rho A(\Omega R)^2} \quad (3)$$

d. Advance Ratio (μ):

$$\mu = \frac{V_T(1.68781)}{\Omega R} \tag{4}$$

Where:

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

SHP_{TR} = Tail rotor output shaft horsepower

ρ = Ambient air density (lb–sec²/ft⁴) $= \rho_o \left[\frac{\delta}{\theta}\right]$

ρ_o = 0.002376892 (lb–sec²/ft⁴)

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 ft²

A_{TR} = Tail rotor disc area = 95 ft²

Ω = Main rotor angular velocity (radians/sec)

Ω_{TR} = Tail rotor angular velocity (radians/sec)

R = Main rotor radius = 26.833 ft

R_{TR} = Tail rotor radius = 5.5 ft

GW = Gross weight (lb)

V_T = True Airspeed (kt) $= \frac{V_E}{1.68781\sqrt{\rho/\rho_o}}$

1.68781 = Conversion factor (ft/sec–kt)

V_E = Equivalent airspeed (ft/sec) =

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

70.7262 = Conversion factor (lb/ft²–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. 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. The torque sensor for both engines was calibrated in a test cell by Corpus Christi Army Depot and the results of this calibration at various engine output shaft speeds are presented in figures C-5 through C-10. 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.

$$SHP_t = \frac{Q(N_P)}{5252.113} \tag{5}$$

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 was determined from the tail rotor's output shaft torque and rotational speed by the following equation:

$$SHPTR = \frac{QTR(NP)(GTR)(0.988036)}{5252.113} \tag{6}$$

Where:

QTR = Tail rotor output shaft torque (ft–lb)

GTR = 15.95825, gear ratio, tail drive shaft to main rotor

0.988036 = Tail rotor gear box efficiency

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.

$$SHP_{instr\ drag} = \frac{F_e\ (\rho/\rho_O)(V_T)^3}{96254} \tag{7}$$

Where:

F_e = 0.833 ft^2 (estimated)

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

Shaft Horsepower Available

5. The shp available for the T700-GE-700 engine installed in the UH-60A was obtained from data received from U.S. Aviation Systems Command. This data was calculated using the General Electric engine deck number 84127A, dated 6 June 1984 for an engine with the HIRSS installed and a power turbine shaft speed of 20,900 rpm. The single engine shaft horsepower available is assumed to be an average of the left and right engines and is presented in figures D-1 and D-2. These figures represent the T700-GE-700 engine with a modified electronic control unit (ECU) $T_{4.5}$ limiter which allows the engine to operate at higher gas turbine temperatures. The modified ECU is from the series of serial numbers that end with G08.

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. Summary hovering performance was then calculated from this nondimensional plot using the power available from reference 3.

Vertical Climb Performance

7. Vertical climb performance was determined for the condition of 4000 feet Hp, 95°F, gross weight equal to 16993.6 pounds, and a rotor speed of 257.9 rpm. Power available at 95 percent intermediate rated power, was obtained from data received from AVSCOM at the above conditions. Power required was determined from OGE hover performance (fig. E-1). Based upon the power available and power required, a generalized excess power coefficient ($\Delta C_{p\ GEN}$) was determined by the following equation.

$$\Delta C_{p\ GEN} = (C_{pc} - C_{ph})/(C_T{}^3/2)^{0.5} \tag{8}$$

Where:

C_{pc} = Coefficient of power climb (power available)

C_{ph} = Coefficient of power hover (power required)

8. Vertical velocity ratio (Vvr) was obtained from figure D-3. The curve on figure D-3 was based on the government competitive testing (ref 11), airworthiness and flight characteristics test (ref 7), and Sikorsky data. The vertical rate of climb (Vv) was determined using the following equation.

$$V_v = V_{vr}\ (V_{tip}\ (C_T/2)^{0.5}) \tag{9}$$

FIGURE D-1

INTERMEDIATE (30 MINUTE LIMIT) POWER AVAILABLE

UH-60A T700-GE-700 SINGLE ENGINE

20,900 OUTPUT SHAFT RPM ZERO KNOTS TRUE AIRSPEED

NOTE: SHP IS AN AVERAGE FOR LEFT AND RIGHT ENGINES OBTAINED FROM REFERENCE 3 BASED ON THE FOLLOWING INSTALLATION CONDITIONS:

1. ENGINE INLET TEMPERATURE RISE = 0.25 DEG C
2. ENGINE INLET PRESSURE RATIO FROM SIKORSKY PIDS, DATED 11 DECEMBER 1987
3. GAS PRODUCER HORSEPOWER EXTRACTED = ZERO
4. CUSTOMER BLEED AIR = ZERO
5. ENGINE ANTI-ICE OFF
6. FUEL HEATING VALUE = 18,300 BTU/LB
7. EXHAUST LOSSES FROM SIKORSKY PIDS, DATED 11 DECEMBER 1987

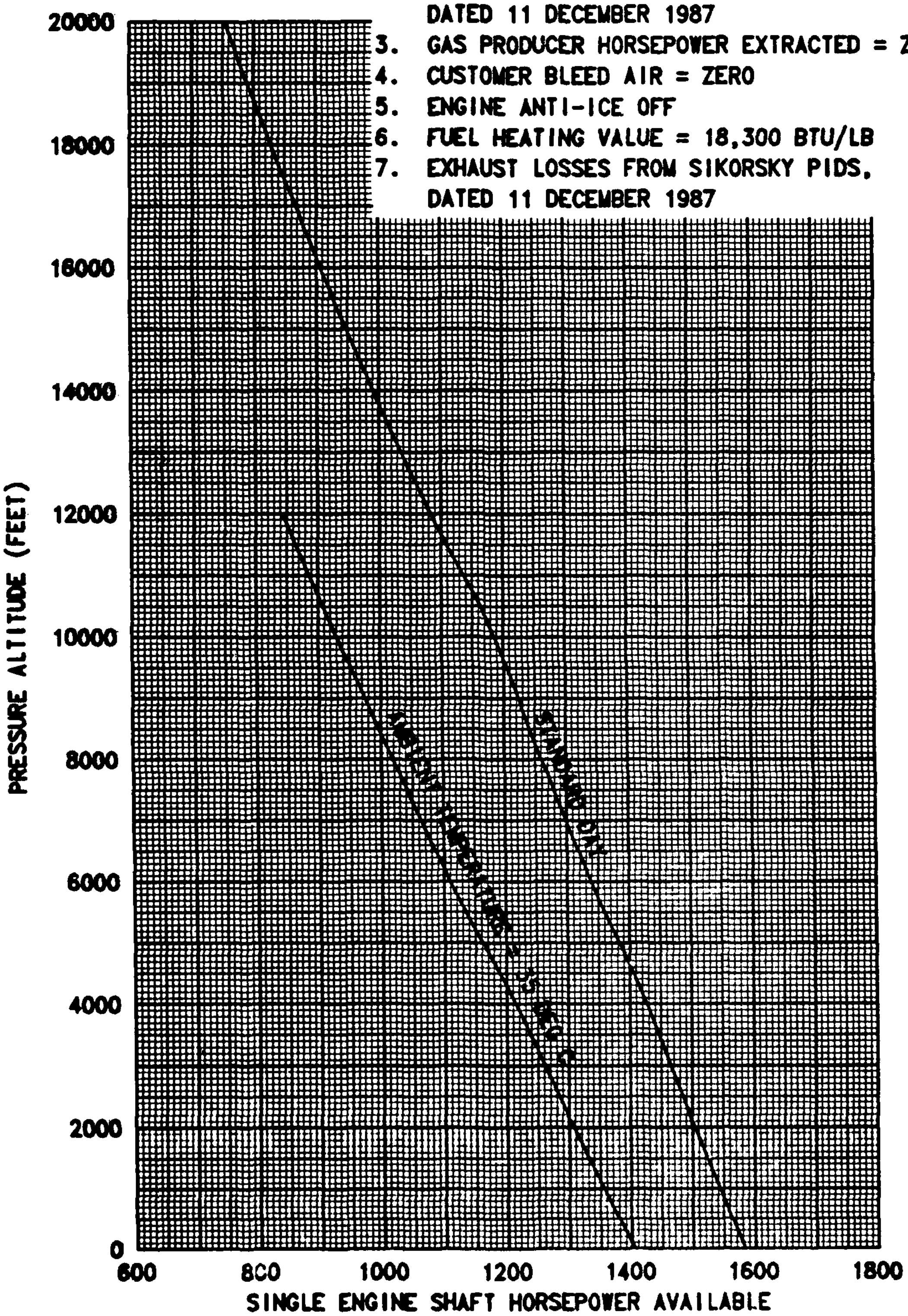


FIGURE D-2

POWER AVAILABLE VERSUS AIRSPEED

UH-60A T700-GE-700 SINGLE ENGINE

20,900 OUTPUT SHAFT RPM

NOTE: SHP IS AN AVERAGE FOR LEFT AND RIGHT ENGINES OBTAINED FROM REFERENCE 3 BASED ON THE FOLLOWING INSTALLATION CONDITIONS:

1. ENGINE INLET TEMPERATURE RISE = 0.25 DEG C
2. ENGINE INLET PRESSURE RATIO FROM SIKORSKY PIDS, DATED 11 DECEMBER 1987
3. GAS PRODUCER HORSEPOWER EXTRACTED = ZERO
4. CUSTOMER BLEED AIR = ZERO
5. ENGINE ANTI-ICE OFF
6. FUEL HEATING VALUE = 18,300 BTU/HR
7. EXHAUST LOSSES FROM SIKORSKY PIDS, DATED 11 DECEMBER 1987

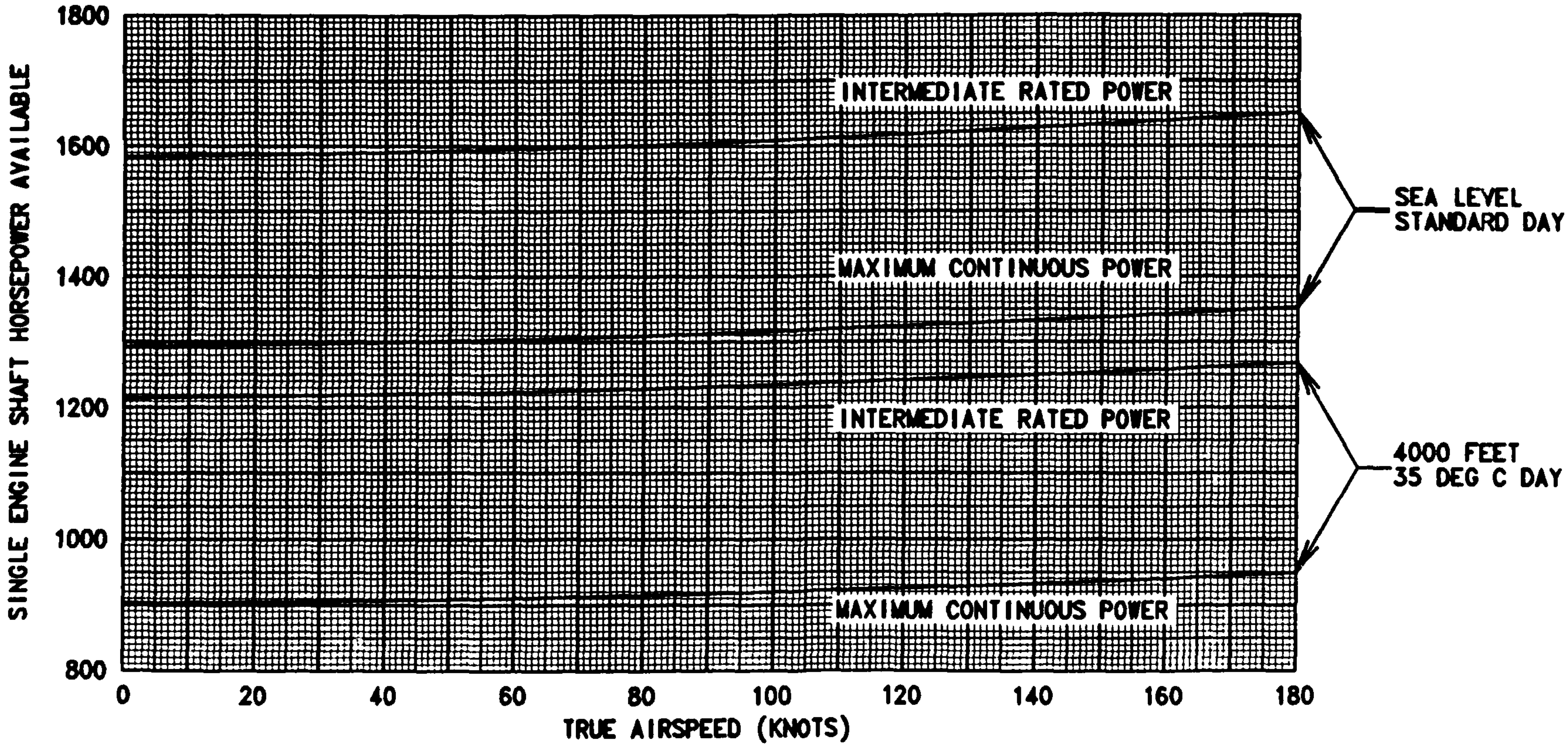
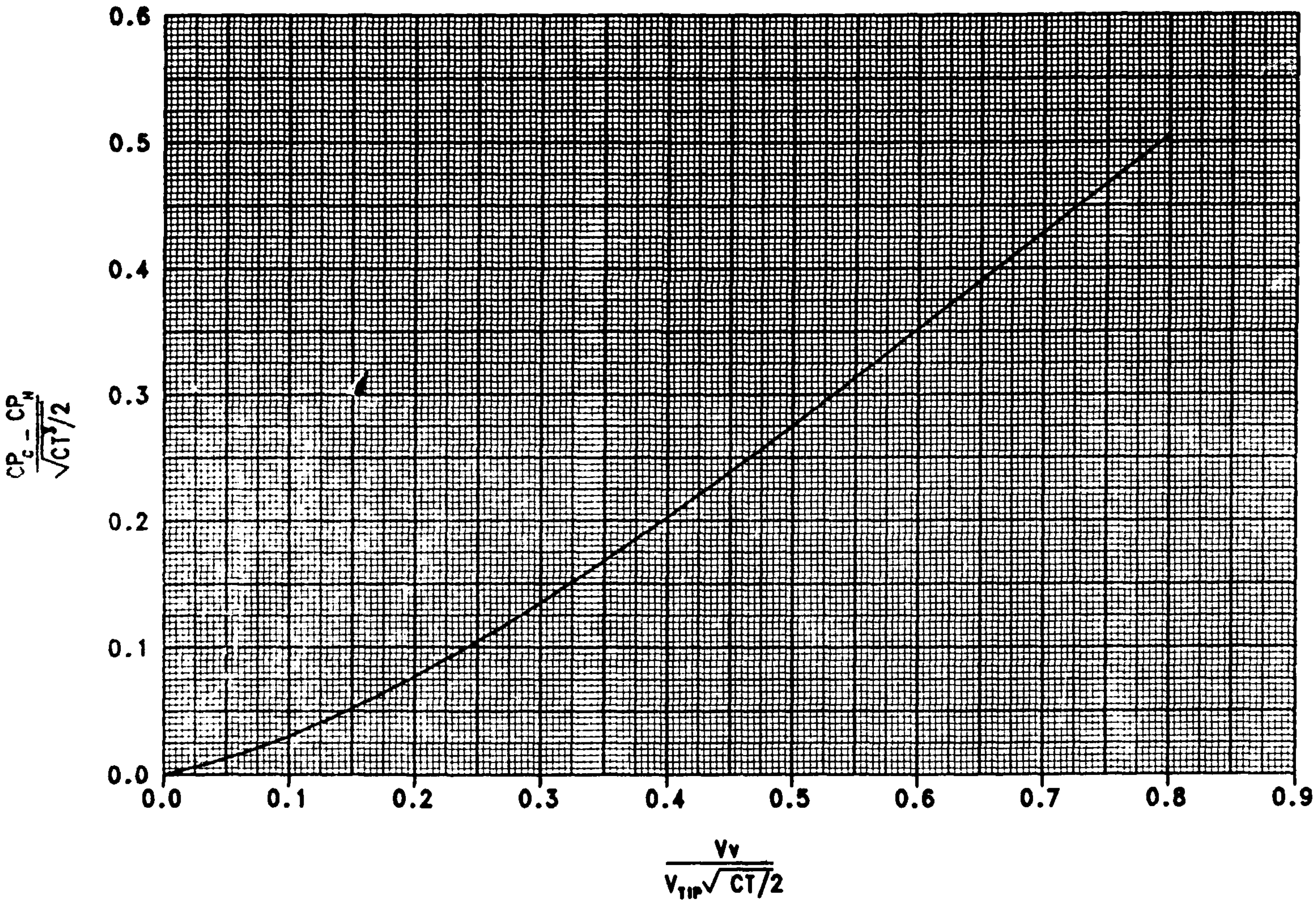


FIGURE D-3

UH-60A, 12th YEAR
VERTICAL CLIMB PERFORMANCE

UH-60A USA S/N 88-26015

CURVE BASED ON AEFA PROJECTS 74-06-1 AND 77-17 AND SIKORSKY DATA



Where:

V_{tip} = Main rotor tip speed (ft/sec)

Level Flight Performance

General:

9. 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.

$$SHP_{R/C} = -\frac{(R/C_{TL})(GW)}{33,000\ (K_p)} \tag{10}$$

$$SHP_{ACCEL} = -1.6098 \times 10^{-4} \left(\frac{\Delta V}{\Delta t}\right)(V_T)(GW) \tag{11}$$

Where:

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

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

OAT_s = Standard ambient temperature at pressure altitude

Where $\frac{\Delta H_P}{\Delta t}$ was measured (°C)

$K_p = 0.76$

1.6098×10^{-4} =Conversion factor (SHP-sec/kt²-lb)

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

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

$$SHP_{CORR} = SHP_t + SHP_{R/C} + SHP_{ACCEL} - SHP_{instr\ drag} - 1.82 \tag{12}$$

10. Each level flight data point was normalized to average conditions for the test by the following equations.

$$SHP_N = SHP_t \frac{(\delta_{AVG}\sqrt{\theta_{AVG}}) \left[\frac{N_R}{\sqrt{\theta}}\right]^3_{AVG}}{(\delta_t\sqrt{\theta_t}) \left[\frac{N_R}{\sqrt{\theta}}\right]^3_t} \quad (13)$$

$$V_{T_N} = V_{T_t} \frac{\left[\frac{N_R}{\sqrt{\theta}}\right]_{AVG}}{\left[\frac{N_R}{\sqrt{\theta}}\right]_t} \quad (14)$$

Where:

N_R = Main rotor speed (rev/min)

subscript t = Measured test data
subscript CORR = Data corrected for test errors
subscript AVG = Average data for the test
subscript N = Data normalized to average conditions

Test data corrected for rate of climb, acceleration, instrumentation installation, standard altitude and ambient temperature are presented in figures E-8 through E-15.

11. Data analysis was accomplished by plotting C_P versus μ for each test at the average C_T and $N_R/\sqrt{\theta}$. These curves were subsequently faired into plots of C_T versus C_P for lines of constant μ at each $N_R/\sqrt{\theta}$ at the average test conditions (figs. E-4 through E-8). These plots allow determination of power required as a function of airspeed for any value of C_T.

12. 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.

$$SHP_{REF} = \frac{SHP_t}{\delta\theta^{0.5}} \quad (15)$$

$$W_{F_{REF}} = \frac{W_{F_t}}{\delta\theta^{0.55}} \quad (16)$$

Where:

W_{F_t} = WFVOL ((FSWI - .006) (55.0 - FUELTMPI))
WFVOL = Fuel Flow Rate (gal/hr)
FSWI = Initial Fuel Specific Weight (lb/gal)
FUELTMPI = Initial Fuel Temperature (°C)
.006 = Specific Weight versus Temperature Gradient for JP-4, JP-5 and JET-A
55.0 = Nominal Fuel Temperature (°C)

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

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

Where:

$\Delta W_F =$ Change in fuel flow between SHP_{CORR} and SHP_N

The following equation was used for determination of SR.

$$SR = \frac{V_{T_N}}{W_{F_N}} \tag{18}$$

APPENDIX E. TEST DATA

FIGURE	FIGURE NUMBER
Hover Performance	E-1 and E-2
Vertical Climb Performance	E-3
Level Flight Performance	E-4 through E-19
Engine Performance	E-20
Ship System Airspeed Calibration	E-21

FIGURE E-1

NONDIMENSIONAL HOVER PERFORMANCE

UH-60A USA S/N 88-26015
100ft WHEEL HEIGHT
NORMAL UTILITY CONFIGURATION

SYMBOL	DENSITY ALTITUDE (FT)	REFERRED ROTOR SPEED (RPM)	OAT (DEG C)
□	3370	246	19.0
⊙	3470	253	20.0
△	3540	261	21.0

NOTES: 1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. VERTICAL DISTANCE FROM BOTTOM OF MAIN WHEELS TO CENTER OF MAIN ROTOR HUB = 12 FT
3. TESTS CONDUCTED WITH THE AIRCRAFT TETHERED TO THE GROUND
4. WINDS LESS THAN THREE KNOTS
5. SHADED SYMBOLS DENOTE FREE FLIGHT HOVER DATA

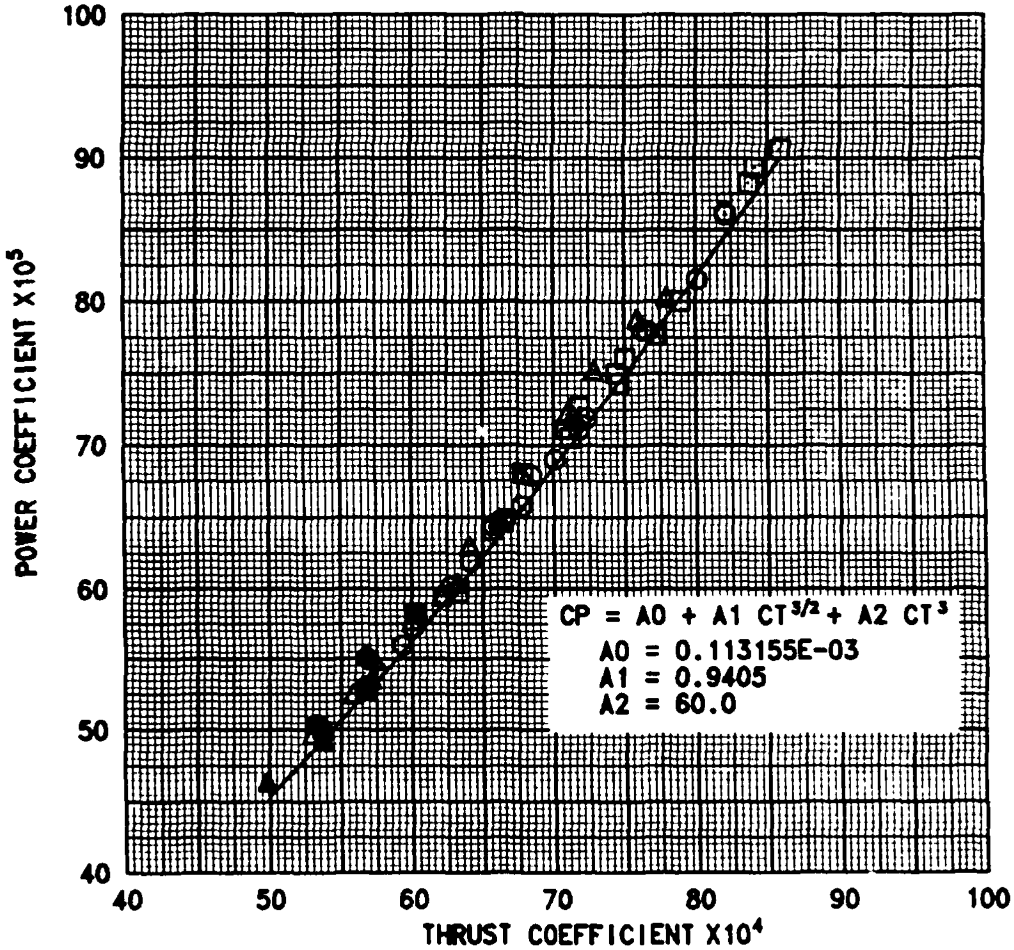


FIGURE E-2

NONDIMENSIONAL HOVER PERFORMANCE

UH-60A USA S/N 88-26015
100ft WHEEL HEIGHT
NORMAL UTILITY CONFIGURATION

SYMBOL	DENSITY ALTITUDE (FT)	REFERRED ROTOR SPEED (RPM)	OAT (DEG C)
□	3370	246	19.0
○	3470	253	20.0
△	3540	261	21.0

NOTES:
1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. VERTICAL DISTANCE FROM BOTTOM OF MAIN WHEELS TO CENTER OF MAIN ROTOR HUB = 12 FT
3. TESTS CONDUCTED WITH THE AIRCRAFT TETHERED TO THE GROUND
4. WINDS LESS THAN THREE KNOTS
5. SHADED SYMBOLS DENOTE FREE FLIGHT HOVER DATA

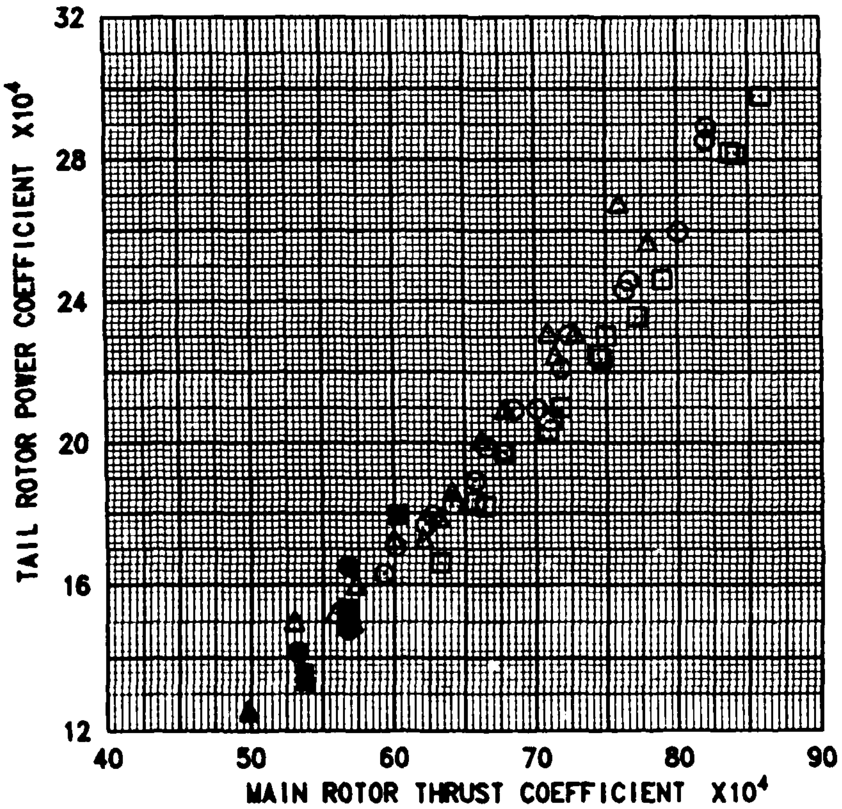


FIGURE E-3

VERTICAL CLIMB PERFORMANCE

UH-60A USA S/N 88-26015

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	PRESSURE ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	THRUST COEFF.	CONFIGURATION
16993.6	347.3(FWD)	0.0	4000.	35.0	257.9	0.007452	NORM UTIL

NOTES: 1. HEADER VALUES DENOTE PIDS GUARANTEE CONDITIONS
2. BALL CENTERED TRIM CONDITION

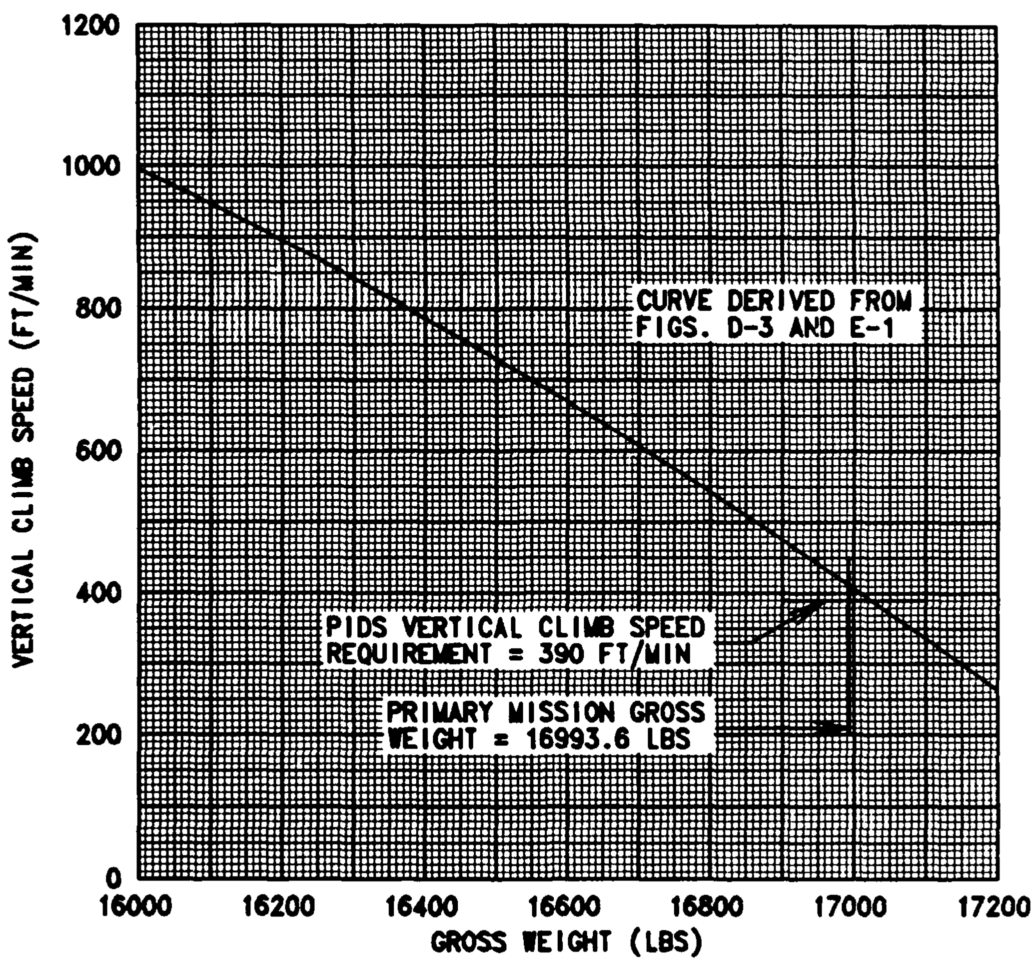


FIGURE E-4
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015
$N_r/\sqrt{\theta}$ = 249.7 RPM

NOTES:
1. NORMAL UTILITY CONFIGURATION
2. BALL CENTERED TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.3
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. POINTS DERIVED FROM FIGURES E-9 THRU E-11

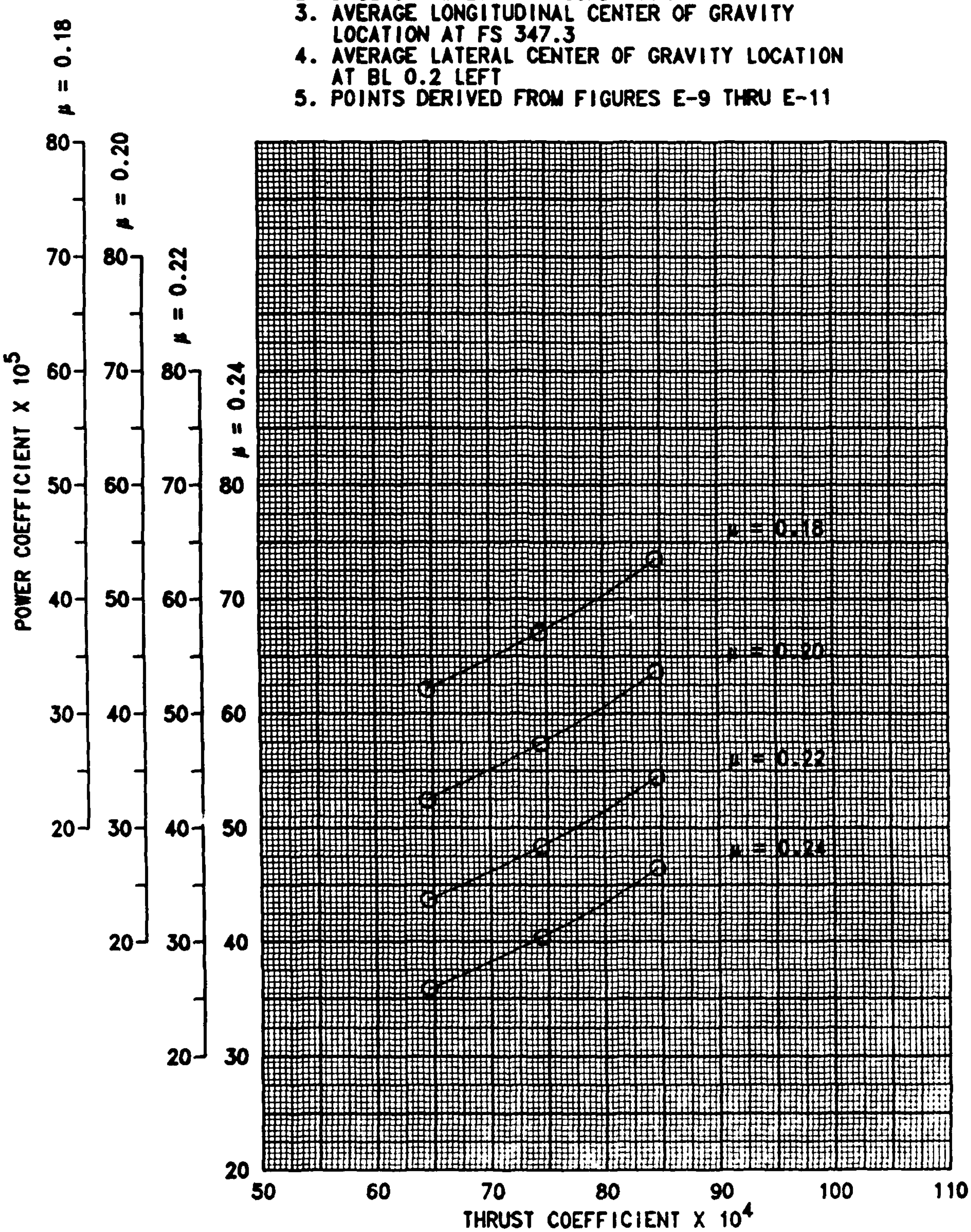


FIGURE E-5

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 88-26015

$N_r/\sqrt{\theta}$ = 249.7 RPM

NOTES:
1. NORMAL UTILITY CONFIGURATION
2. BALL CENTERED TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.3
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.2 LEFT
5. POINTS DERIVED FROM FIGURES E-9 THRU E-11

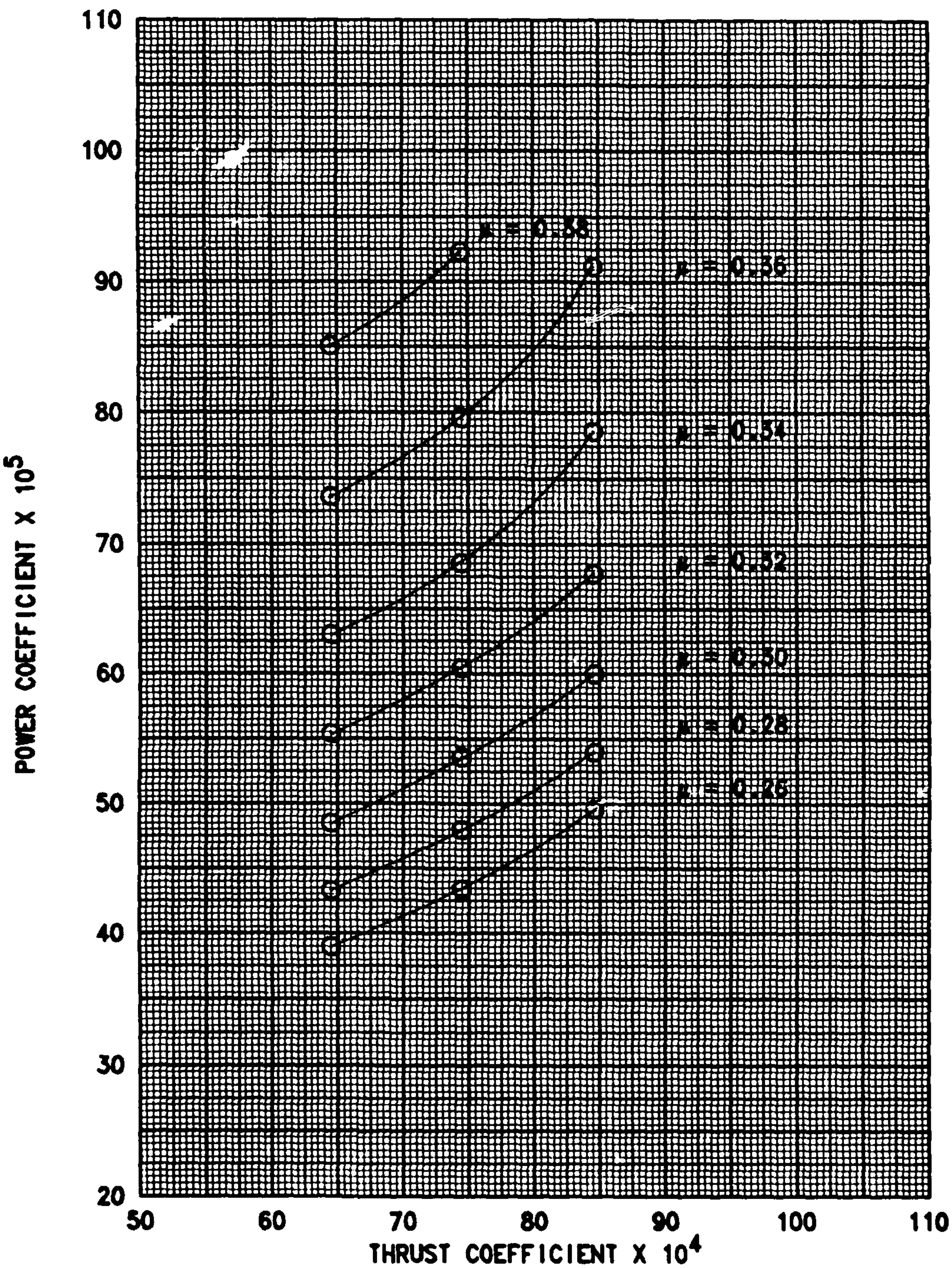


FIGURE E-6
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015
$N_r/\sqrt{\theta}$ = 258.3 RPM

NOTES:
1. NORMAL UTILITY CONFIGURATION
2. BALL CENTERED TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.6
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.0 MID
5. POINTS DERIVED FROM FIGURES E-12 THRU E-16

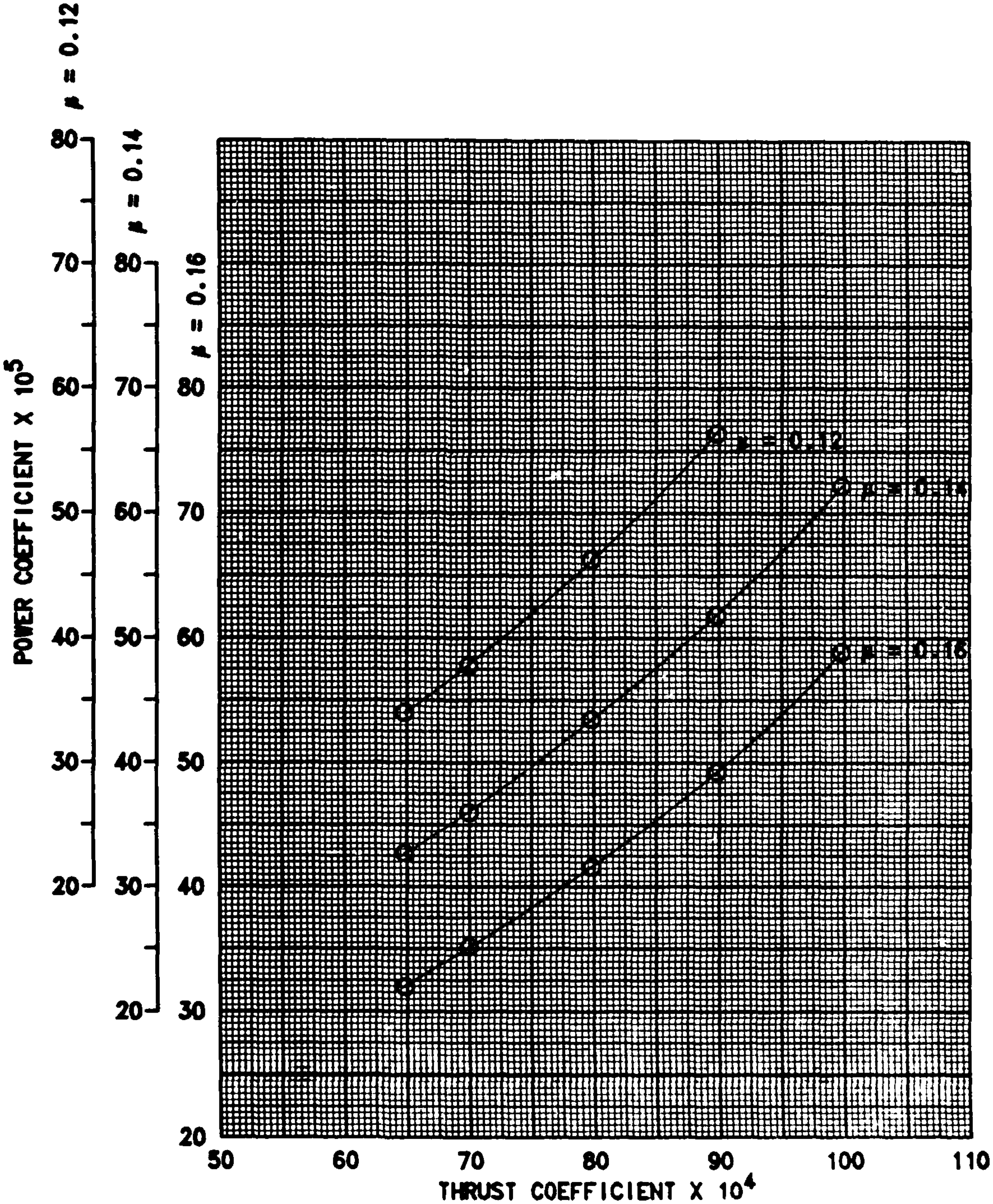


FIGURE E-7

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 88-26015

$N_r/\sqrt{\theta}$ = 258.3 RPM

NOTES:
1. NORMAL UTILITY CONFIGURATION
2. BALL CENTERED TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.6
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.0 MID
5. POINTS DERIVED FROM FIGURES E-12 THRU E-16

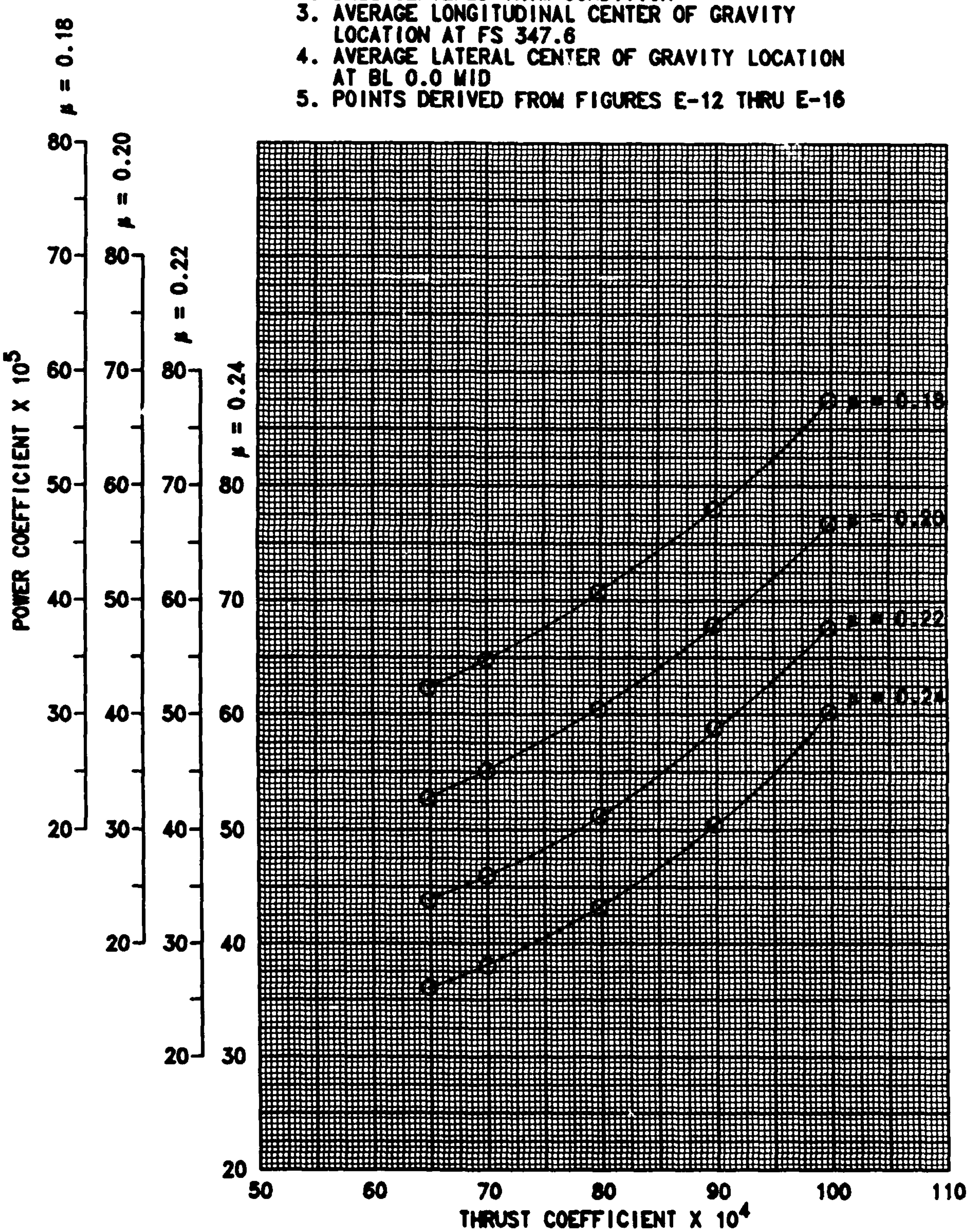


FIGURE E-8
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015
$N_r/\sqrt{\theta}$ = 258.3 RPM

NOTES:
1. NORMAL UTILITY CONFIGURATION
2. BALL CENTERED TRIM CONDITION
3. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 347.6
4. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.0 MID
5. POINTS DERIVED FROM FIGURES E-12 THRU E-16

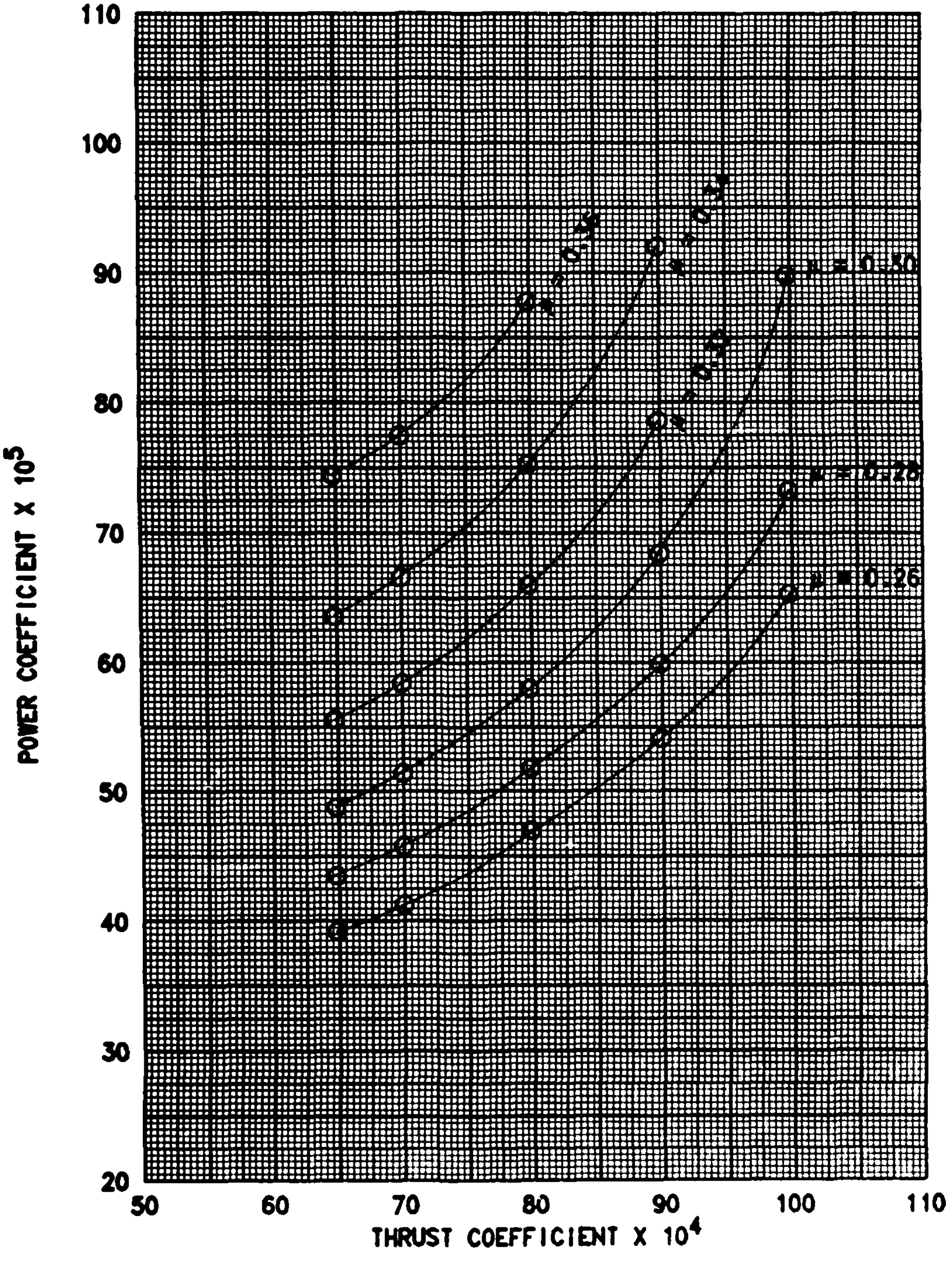


FIGURE E-9
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
15810	347.0(FWD)	0.0	4200	28.5	249.9	0.006453	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

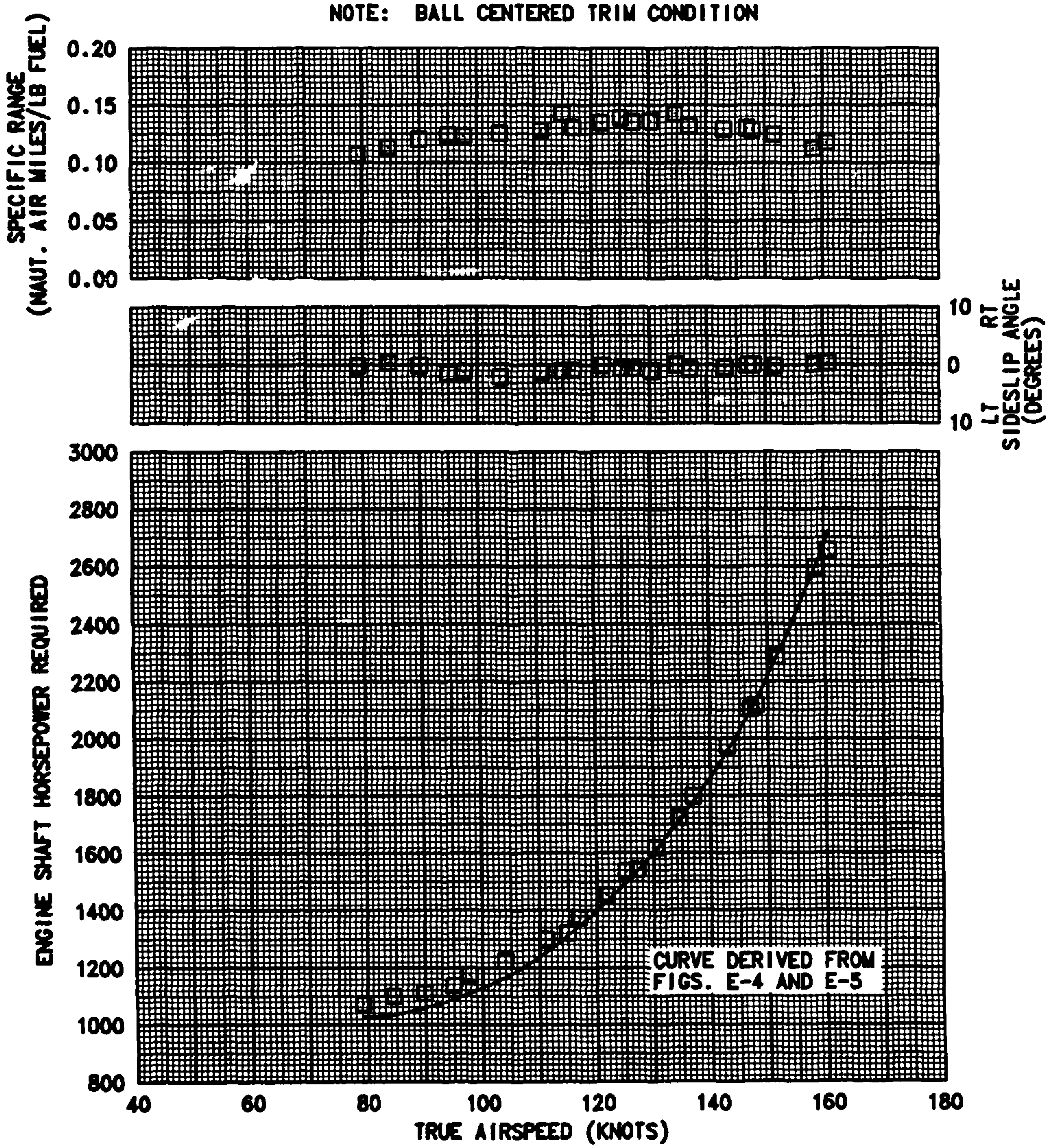


FIGURE E-10
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
16290	347.2(FWD)	0.5LT	7500	26.0	249.4	0.007444	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

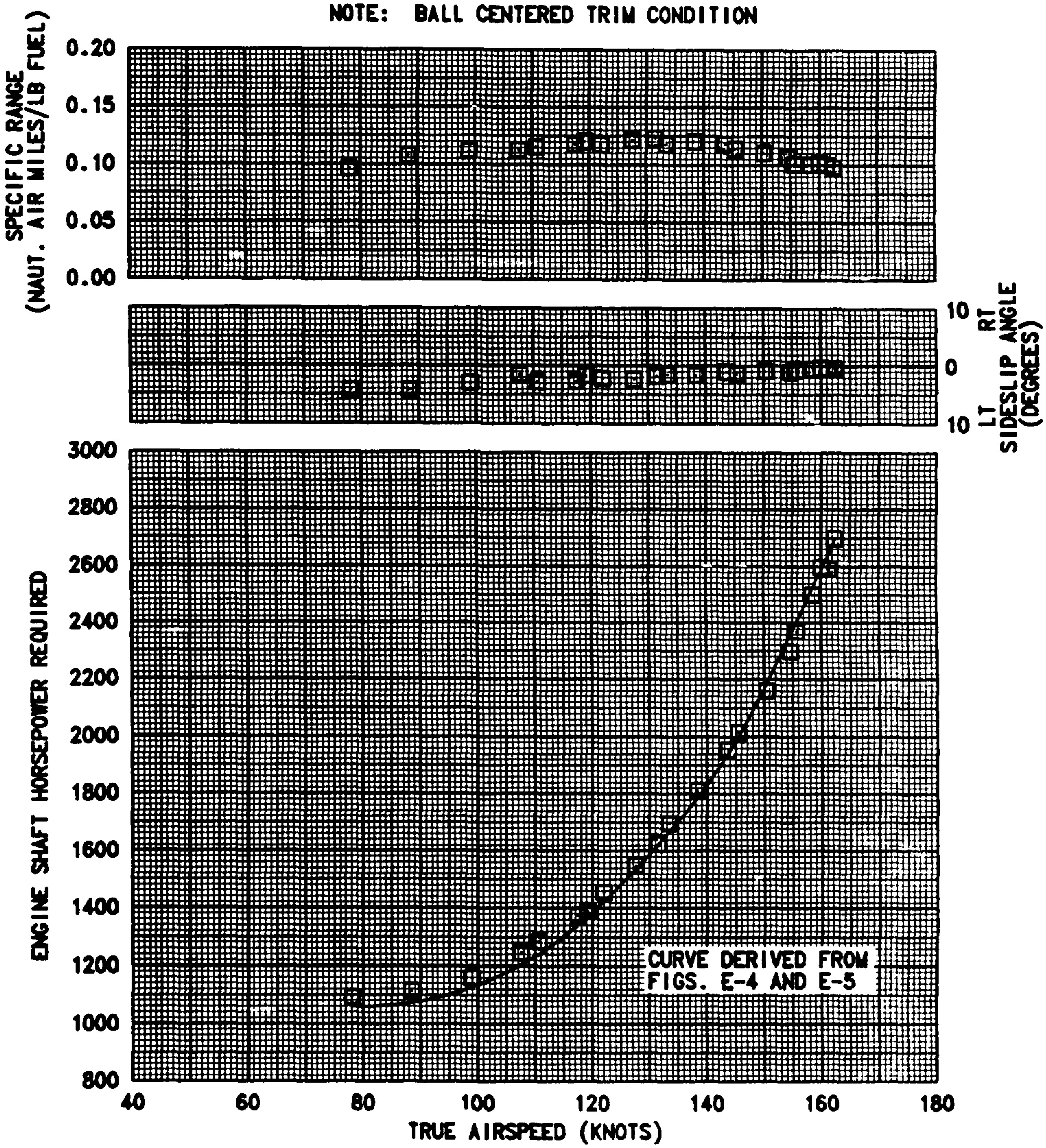


FIGURE E-11
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
16520	347.3(FWD)	0.0	9990	14.5	249.8	0.008457	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

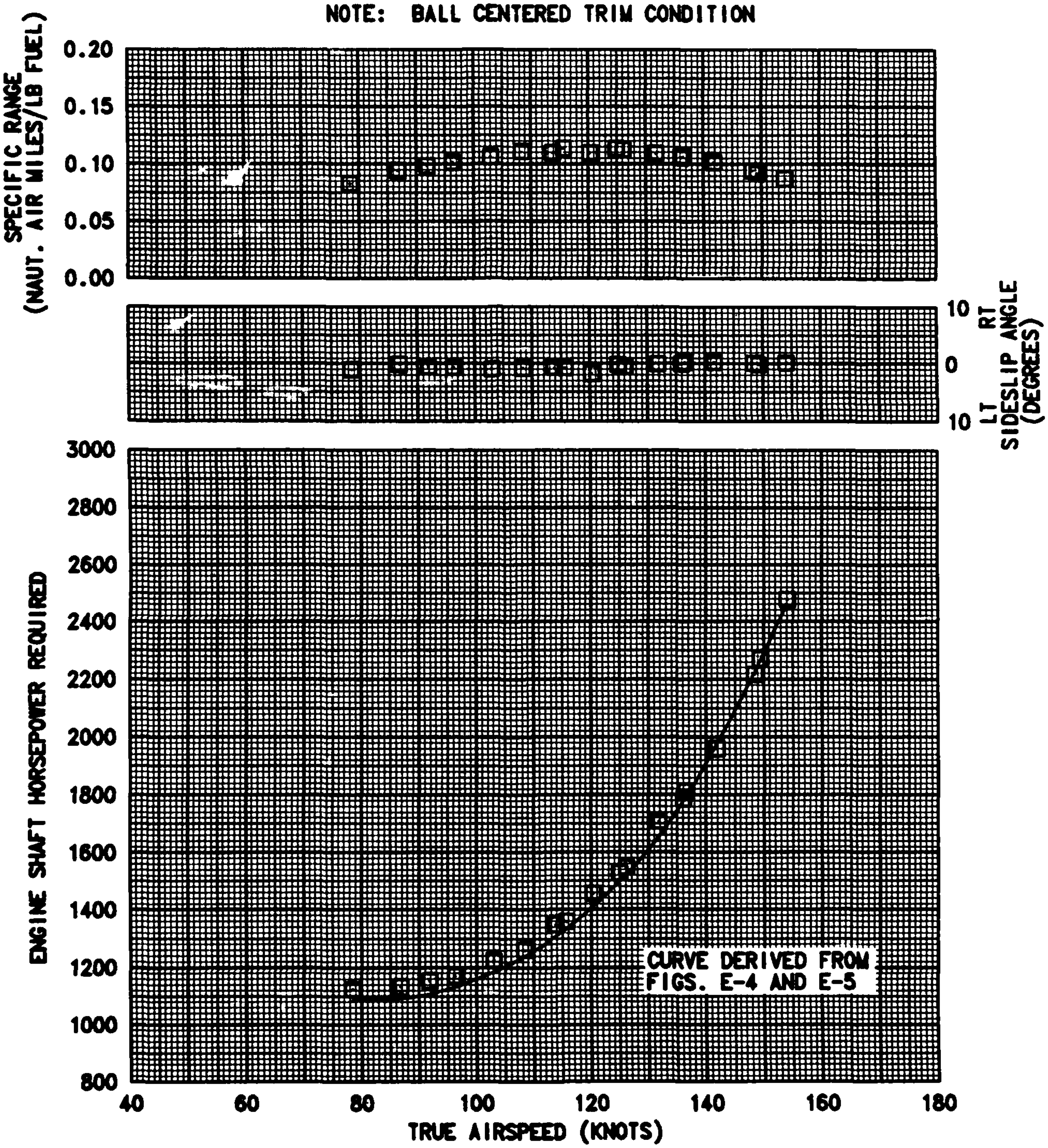


FIGURE E-12
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
16480	347.9(FWD)	0.0	4270	20.5	258.3	0.006482	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

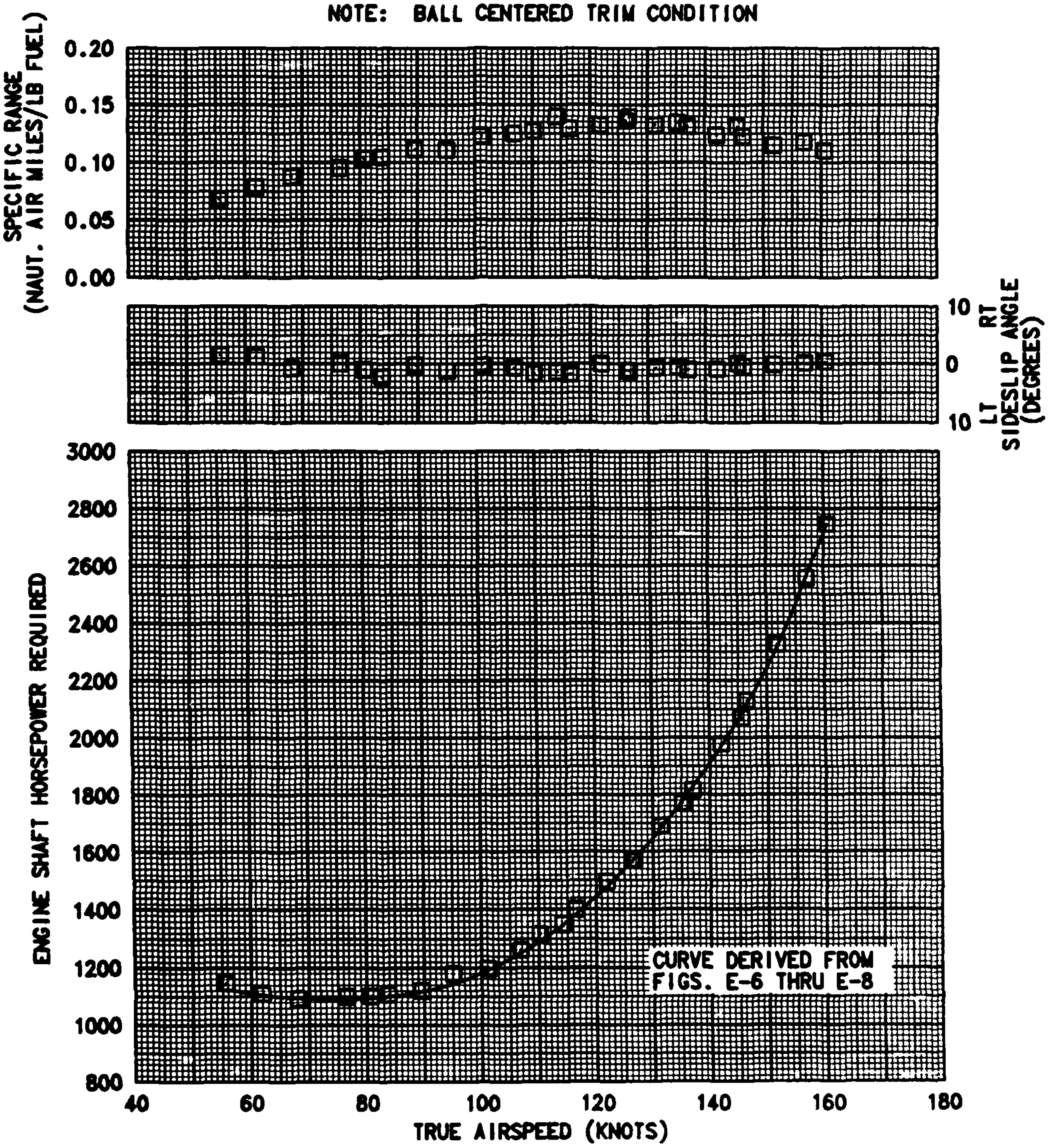


FIGURE E-13
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
16840	347.8(FWD)	0.0	6520	21.0	258.3	0.006996	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

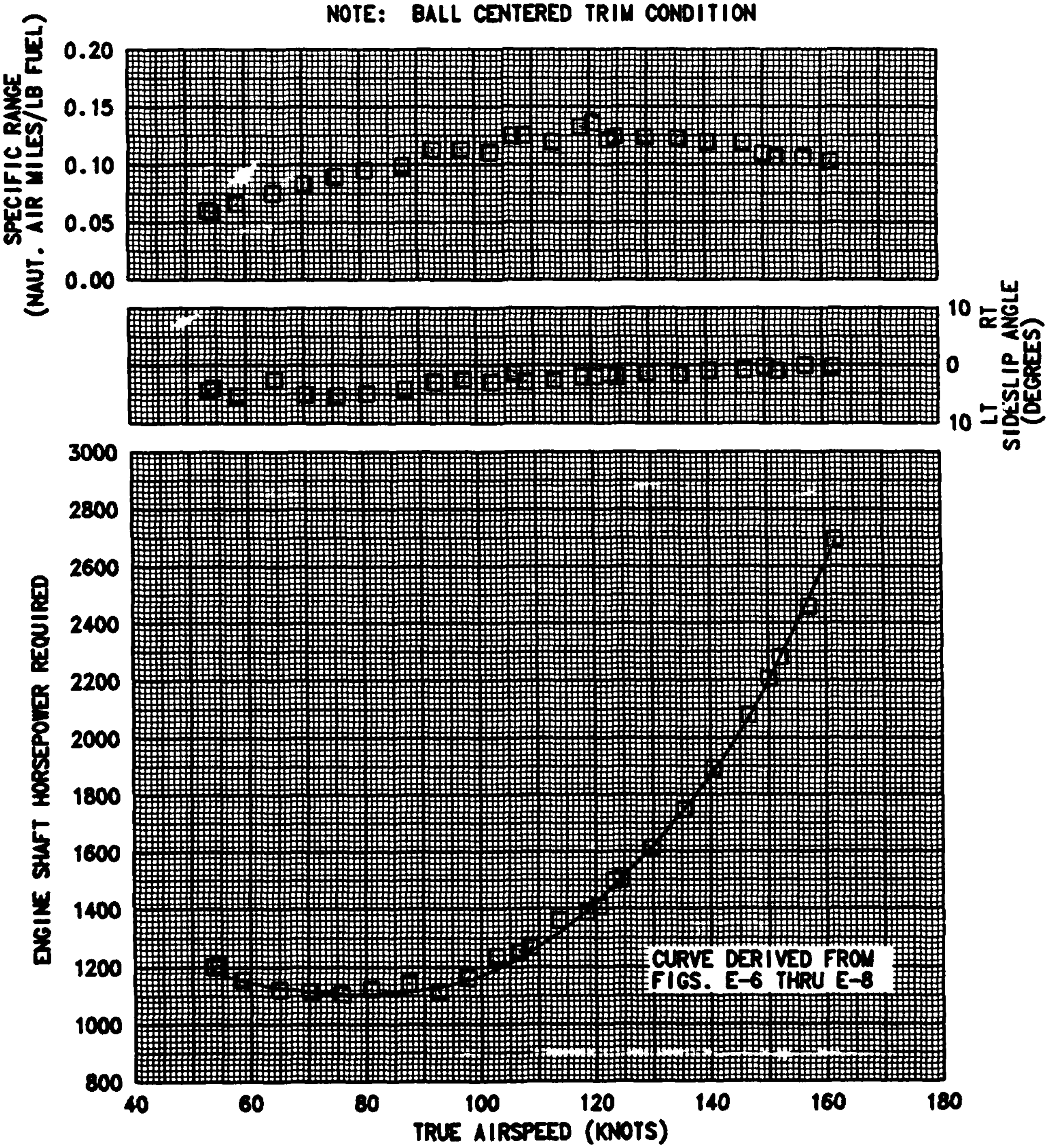


FIGURE E-14

LEVEL FLIGHT PERFORMANCE

UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
18400	347.6(FWD)	0.0	7560	21.0	258.4	0.007981	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

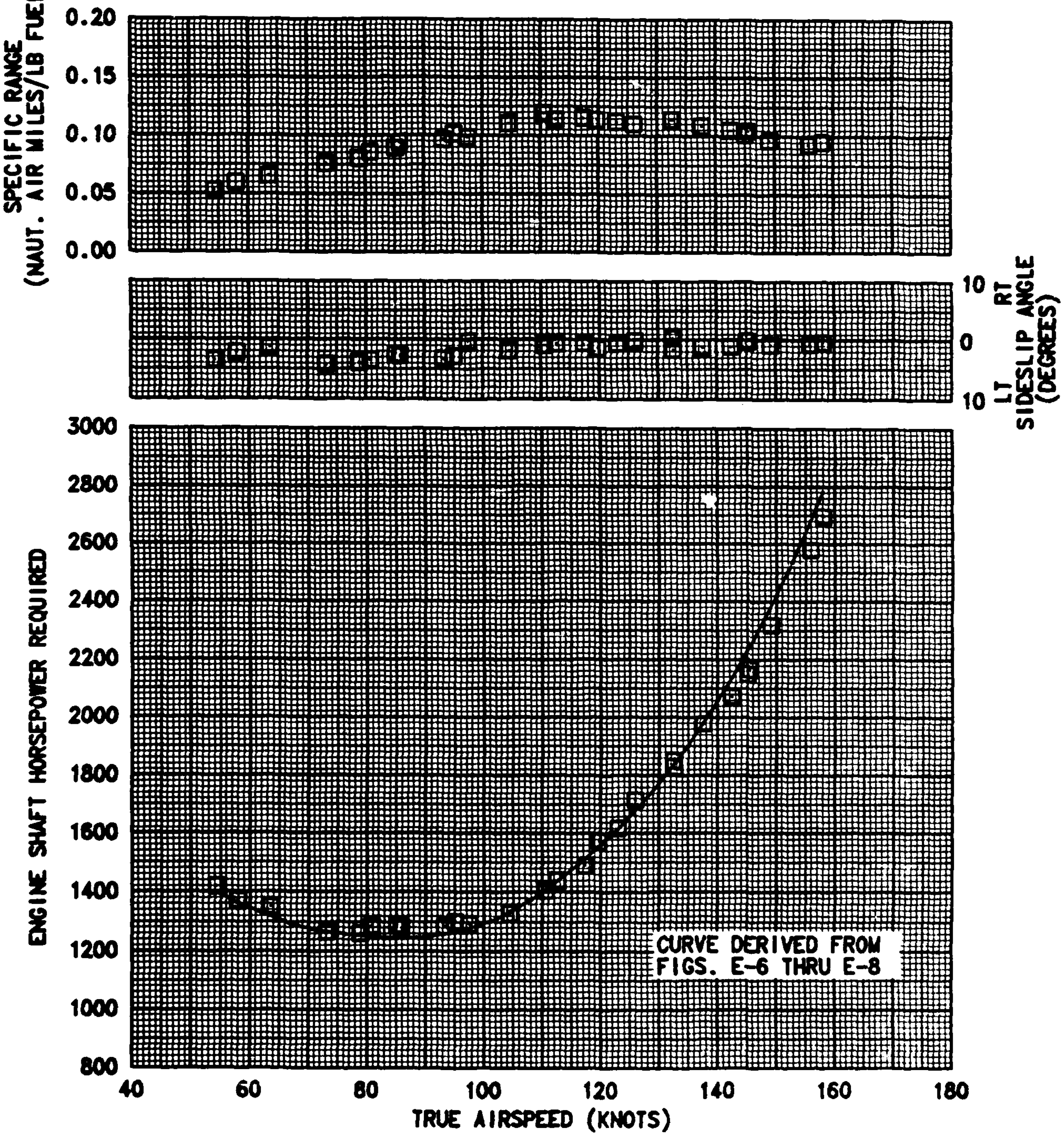


FIGURE E-15
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
18580	347.6(FWD)	0.0	10250	13.5	258.5	0.008986	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

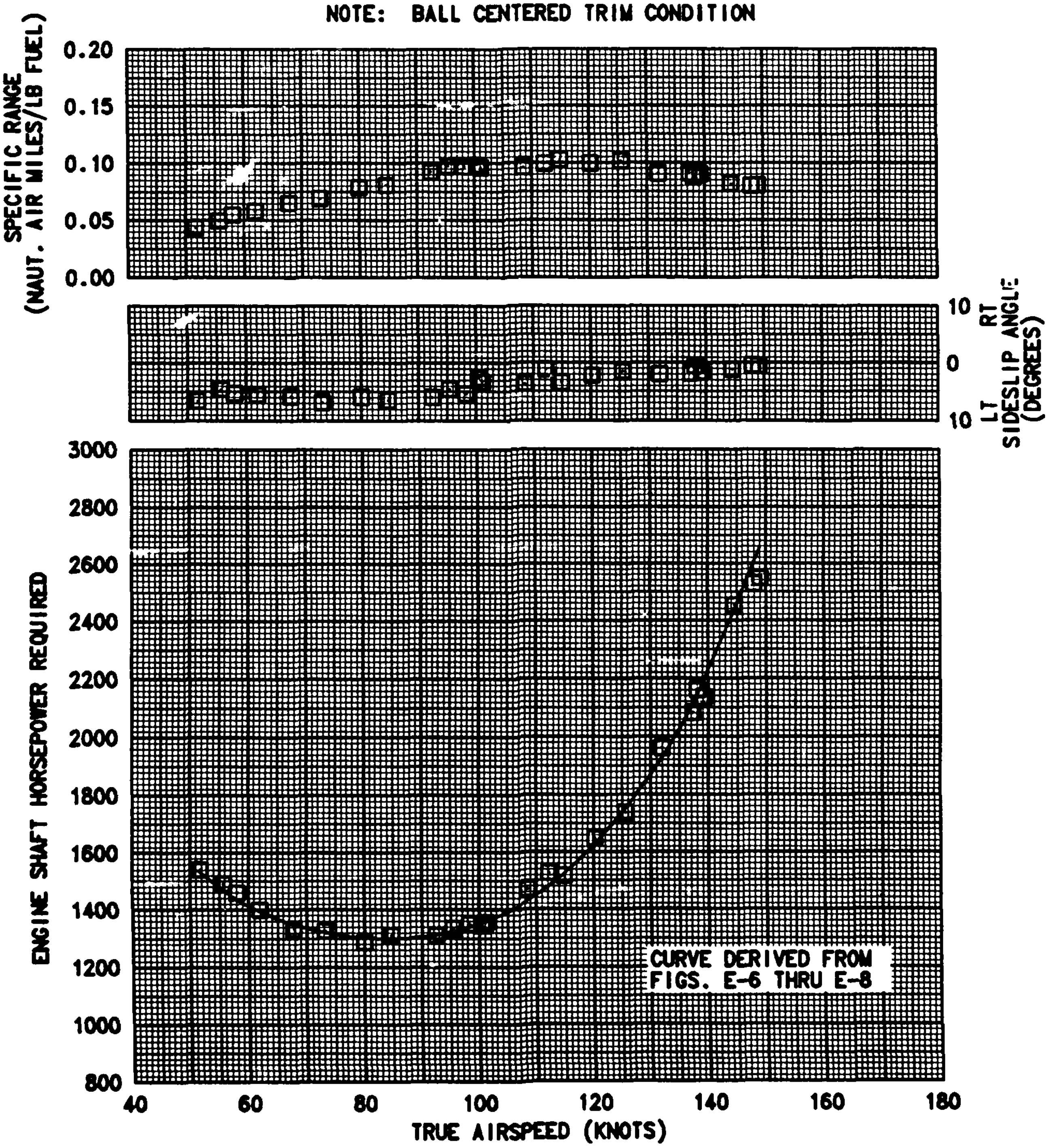


FIGURE E-16
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
20330	347.3(FWD)	0.0	10730	14.5	258.1	0.009979	NORM UTIL

NOTE: BALL CENTERED TRIM CONDITION

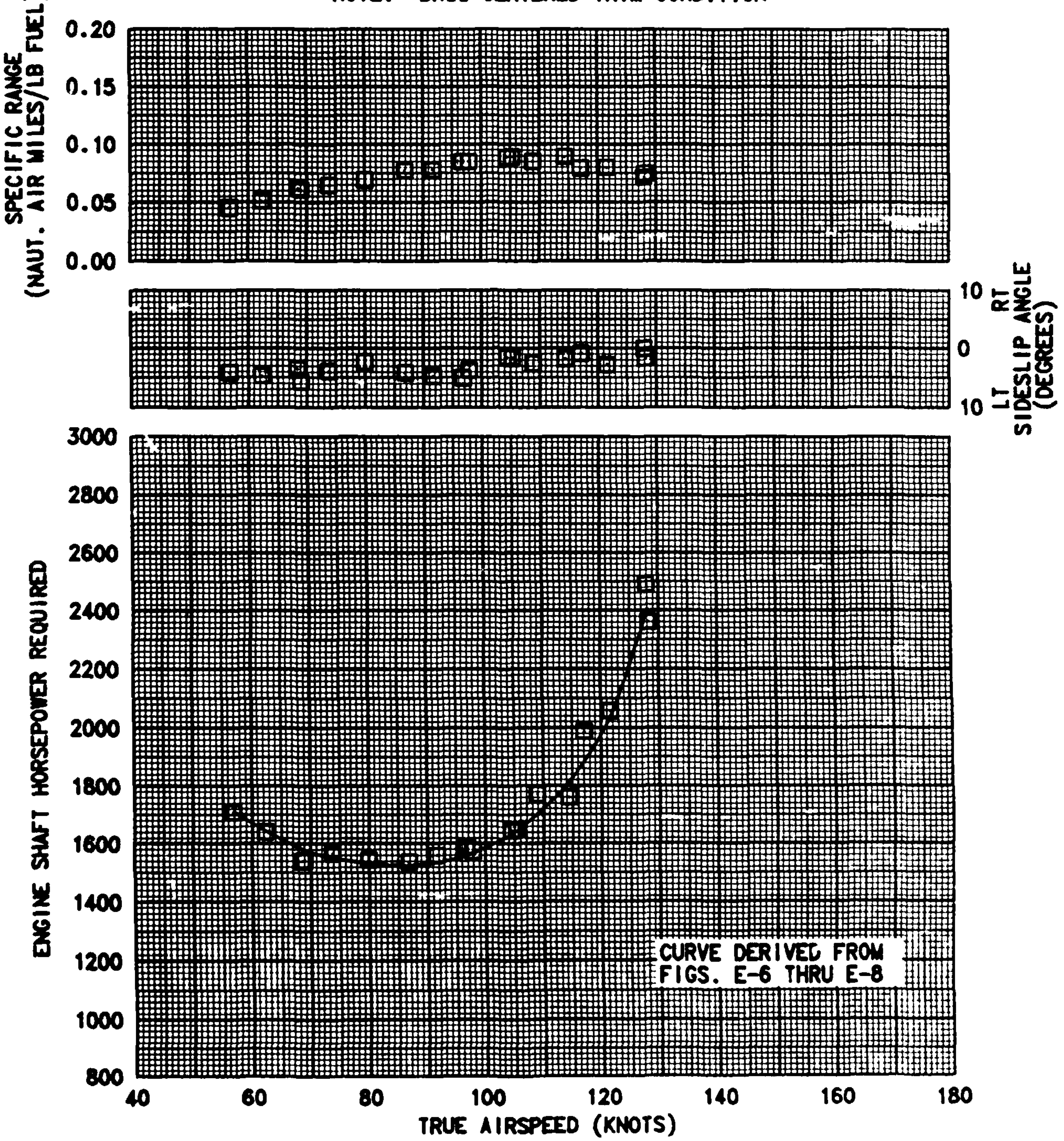


FIGURE E-17
LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 88-26015

GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	PRESSURE ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	THRUST COEFF.	CONFIGURATION
16993.6	347.2(FWD)	0.0	4000	35.0	257.9	0.007452	NORM UTIL

NOTES: 1. HEADER VALUES DENOTE PIDS GUARANTEE CONDITIONS
2. BALL CENTERED TRIM CONDITION

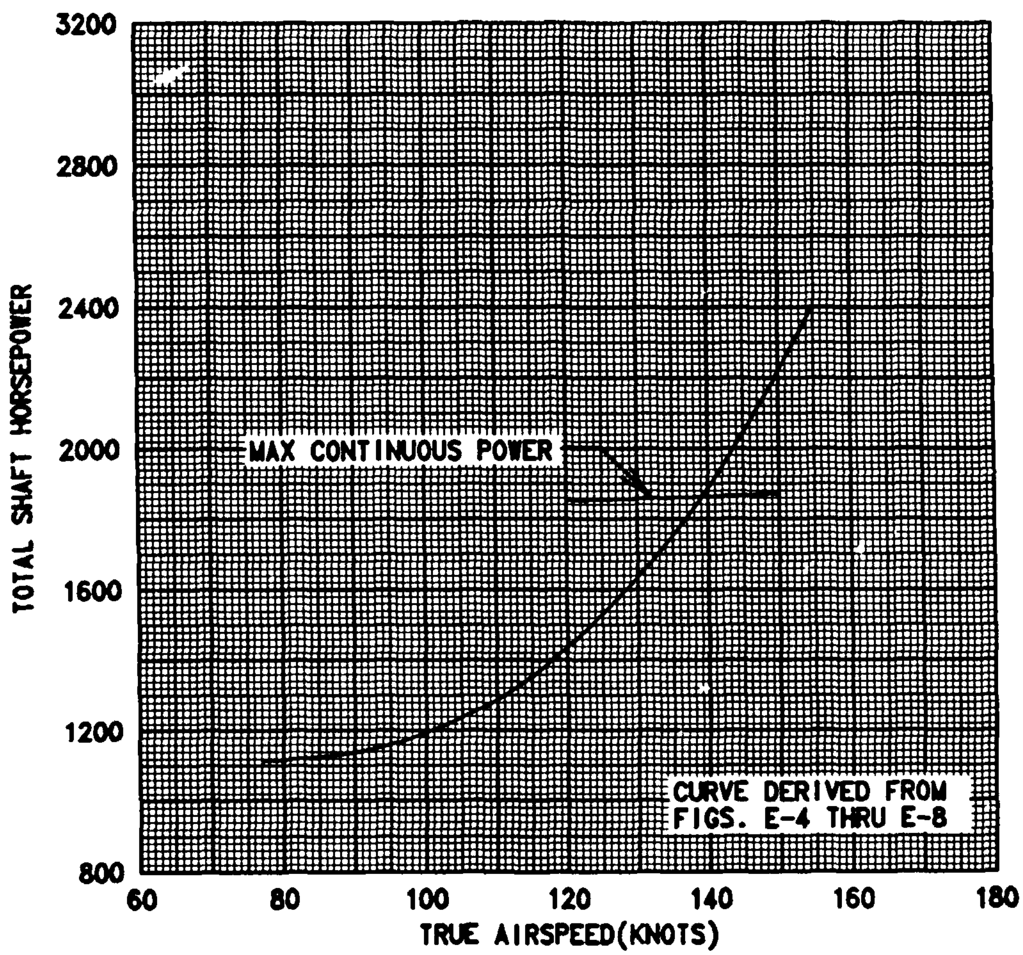


FIGURE E-18

TAIL ROTOR SHAFT HORSEPOWER VS TRUE AIRSPEED

UH-60A USA S/N 88-26015

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT	CONFIGURATION
□	15810	347.0(FWD)	0.0	4200	28.5	249.9	0.008453	NORMAL UTILITY
○	16290	347.2(FWD)	0.5LT	7500	26.0	249.4	0.007444	NORMAL UTILITY
△	16520	347.8(FWD)	0.0	9660	14.5	249.8	0.008457	NORMAL UTILITY

NOTE: BALL CENTERED FLIGHT

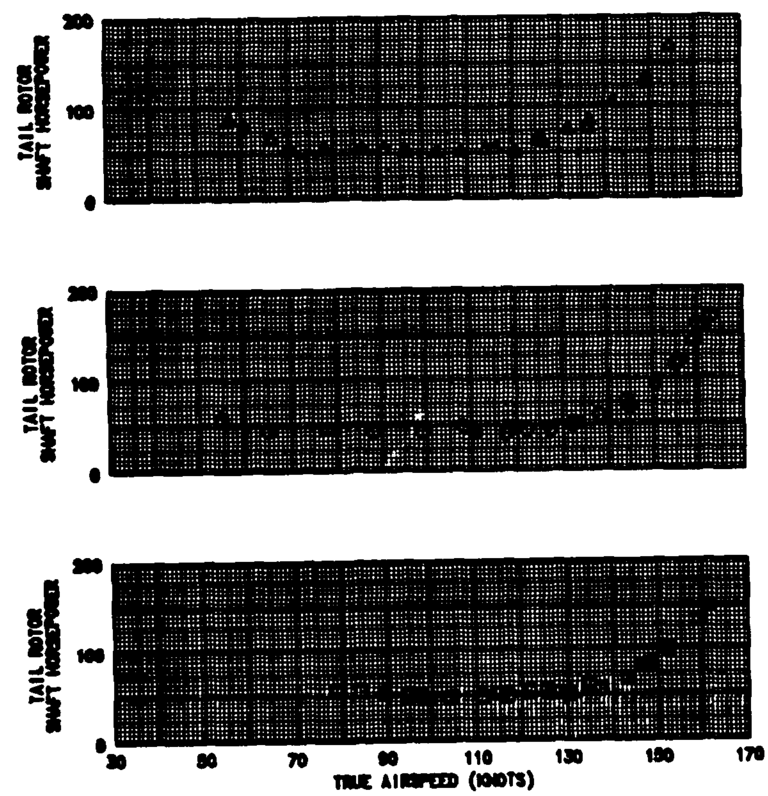


FIGURE E-19
TAIL ROTOR SHAFT HORSEPOWER VS TRUE AIRSPEED
UH-60A USA S/N 88-25015

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG THRUST COEFFUCIENT	CONFIGURATION
⊡	16480	347.9(FWD)	0.0	4270	20.5	258.3	0.006482	NORMAL UTILITY
⊙	16540	347.8(FWD)	0.0	6620	21.0	258.3	0.006996	NORMAL UTILITY
△	18400	347.6(FWD)	0.0	7580	21.0	258.4	0.007981	NORMAL UTILITY
+	18580	347.6(FWD)	0.0	10250	13.5	258.5	0.008886	NORMAL UTILITY
◆	20330	347.3(MID)	0.0	10370	14.5	258.1	0.008979	NORMAL UTILITY

NOTE: BALL CENTERED FLIGHT

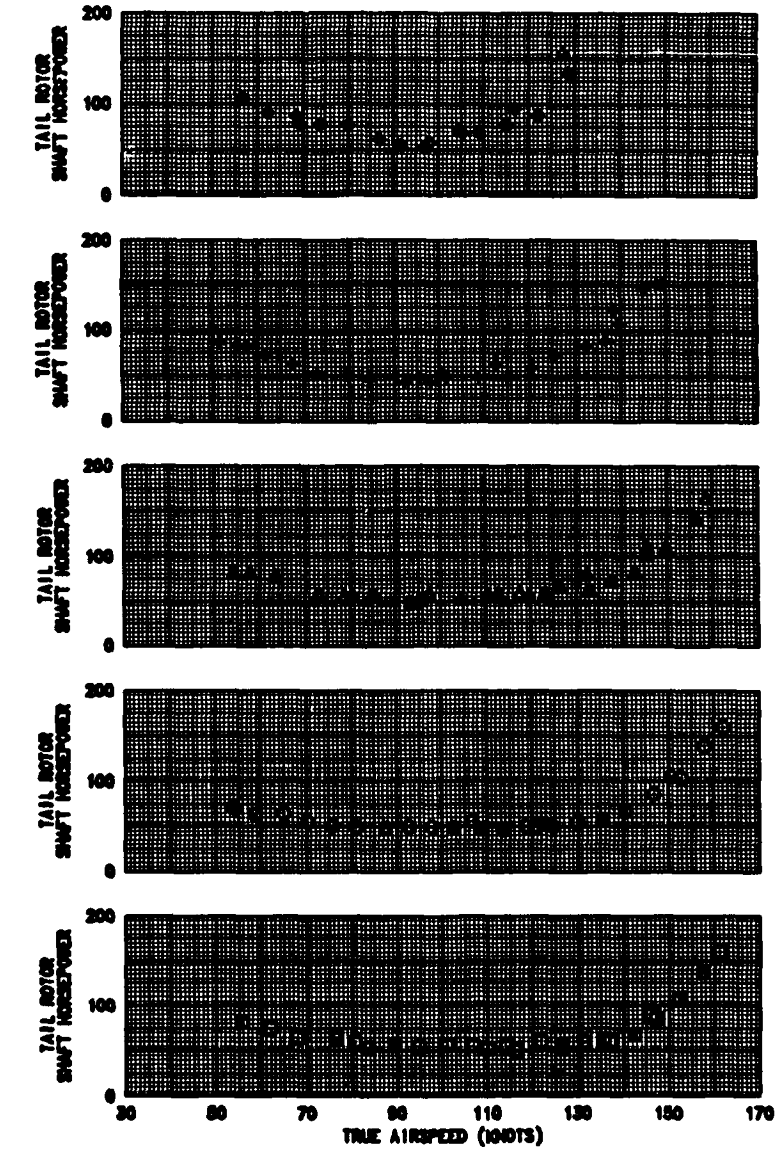


FIGURE E-20

TOTAL ENGINE POWER VS. TOTAL ROTOR POWER

UH-60A USA S/N 88-26015

NORMAL UTILITY CONFIGURATION

SYMBOL	DENSITY ALTITUDE (FT)	REFERRED ROTOR SPEED (RPM)	OAT (DEG C)	FLIGHT CONFIGURATION
⊡	3470	246, 253, 261	20.0	HOVER
⌑	4200	249.9	28.5	LEVEL, CT=64.53
⊞	7500	249.4	26.0	LEVEL, CT=74.44
+	9990	249.8	14.5	LEVEL, CT=84.57
⊙	4270	258.3	20.5	LEVEL, CT=64.82
△	6520	258.3	21.0	LEVEL, CT=69.96
◇	7560	258.4	21.0	LEVEL, CT=79.81
⊠	10250	258.5	13.5	LEVEL, CT=89.86
⊕	10730	258.1	14.5	LEVEL, CT=99.79

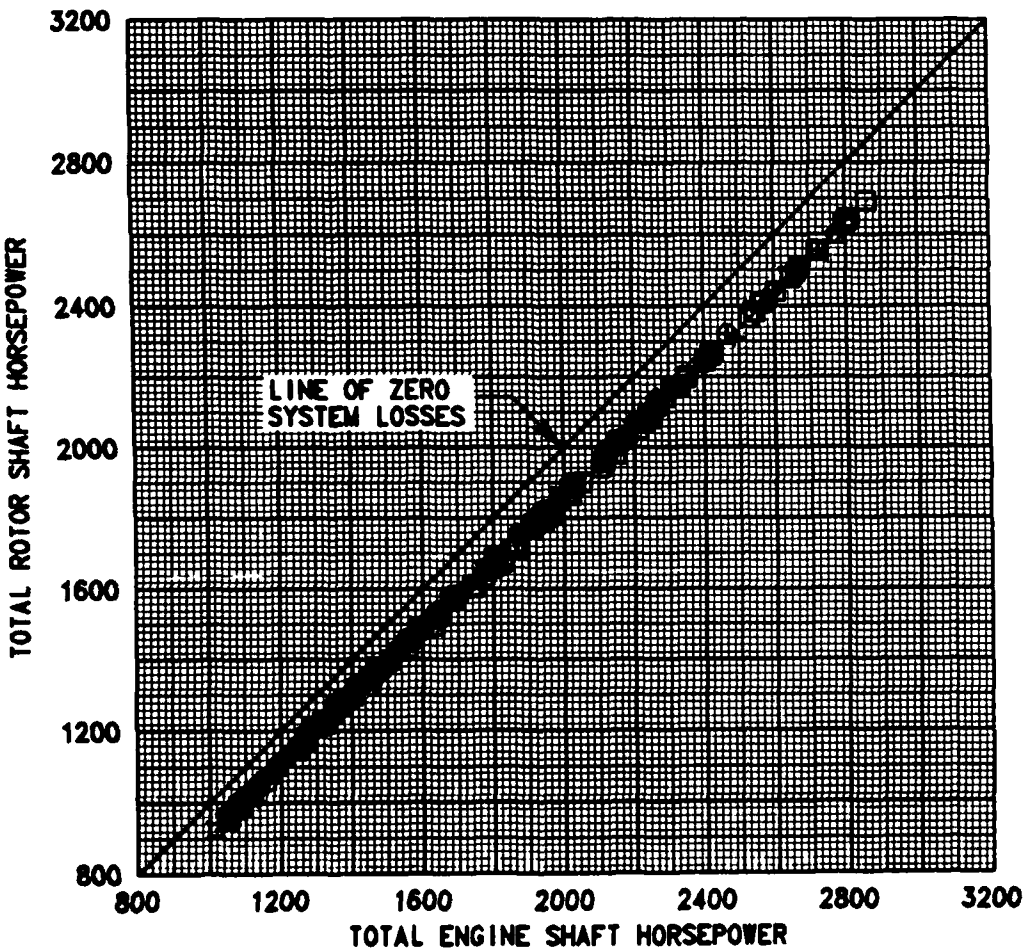


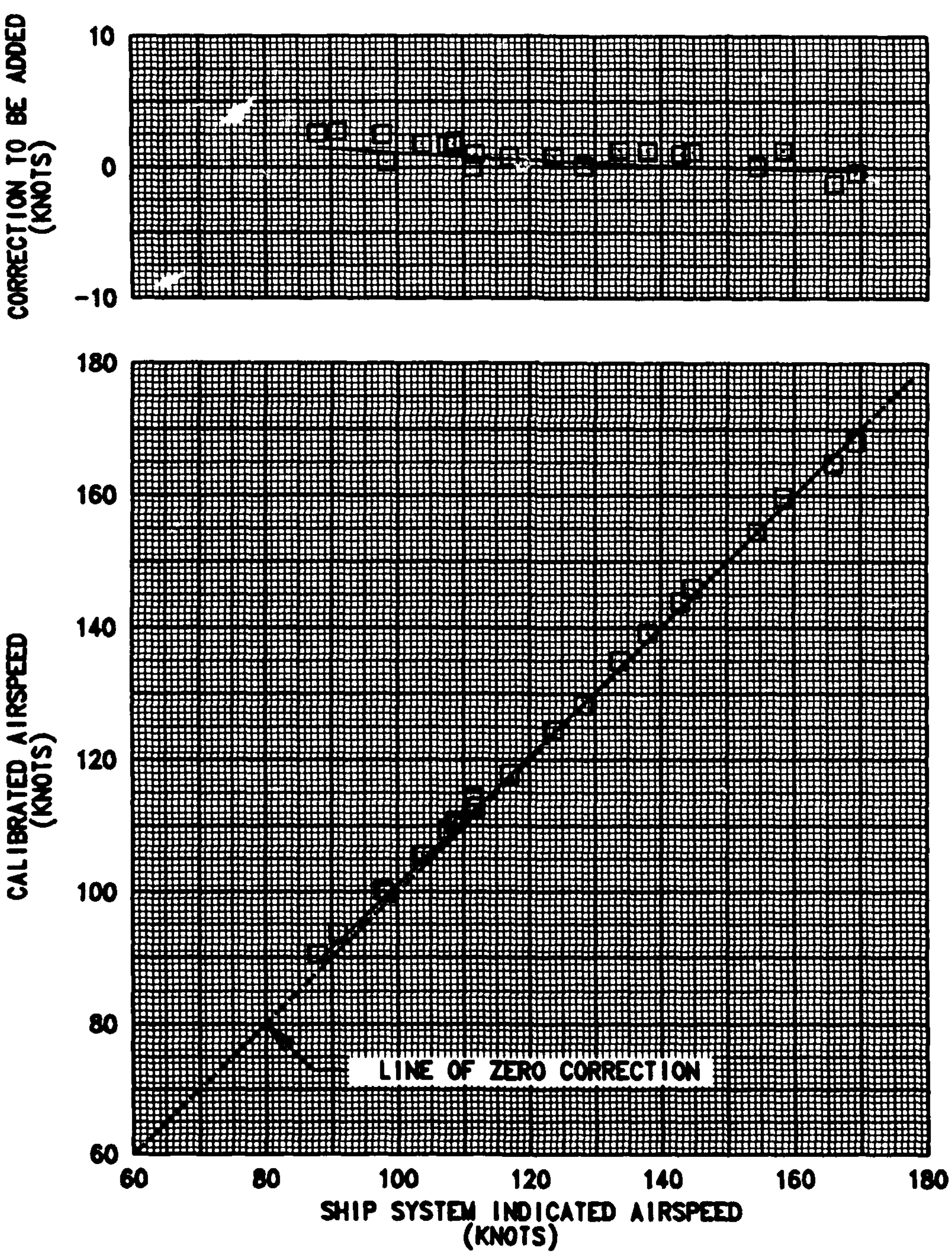
FIGURE E-21

SHIP SYSTEM AIRSPEED CALIBRATION IN LEVEL FLIGHT

UH-60A USA S/N 88-26015

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
16710	347.6(FWD)	0.0	8410	23.5	258	T-34C PACE

NOTES: 1. NORMAL UTILITY CONFIGURATION
2. BALL CENTERED FLIGHT



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