AD A132068

(12)

USAAEFA PROJECT NO. 77-17

AIRWORTHINESS AND FLIGHT CHARACTERISTICS EVALUATION UH-60A (BLACK HAWK) HELICOPTER

FINAL REPORT

JOHN I. NAGATA
PROJECT OFFICER

GARY L. SKINNER
PROJECT ENGINEER

ROBERT M. BUCKANIN
PROJECT ENGINEER

ROBERT D. ROBBINS
MAJ, TC
PROJECT PILOT

ROBERT A. WILLIAMS
CW4, AV
PROJECT PILOT

SEPTEMBER 1981

SEP 2 1983
A

Approved for public release; distribution unlimited.

UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

DTIC FILE COPY

83 09 01 004

DISCLAIMER NOTICE

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

DISPOSITION INSTRUCTIONS

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

TRADE NAMES

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

UNCLASSIFIED
SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

REPORT DOCUMENTATION PAGE	READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER: USAAEFA ~~PROJECT NO.~~ 77-17	2. GOVT ACCESSION NO.: AD A132 066 / 3. RECIPIENT'S CATALOG NUMBER
4. TITLE (and Subtitle): AIRWORTHINESS AND FLIGHT CHARACTERISTICS EVALUATION, UH-60A (BLACK HAWK) HELICOPTER	5. TYPE OF REPORT & PERIOD COVERED: FINAL, 27 OCT 1979 - 9 OCT 1980
	6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s): JOHN I. NAGATA, ROBERT M. BUCKANIN, GARY L. SKINNER, ROBERT D. ROBBINS, ROBERT A. WILLIAMS	8. CONTRACT OR GRANT NUMBER(s)
9. PERFORMING ORGANIZATION NAME AND ADDRESS: US ARMY AVN ENGR FLIGHT ACTIVITY, EDWARDS AIR FORCE BASE, CALIFORNIA	10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: 68-0-BH031-01-68
11. CONTROLLING OFFICE NAME AND ADDRESS: US ARMY AVN RESEARCH & DEVELOPMENT COMMAND, 4300 GOODFELLOW BOULEVARD, ST. LOUIS, MISSOURI	12. REPORT DATE: SEPTEMBER 1981
	13. NUMBER OF PAGES: 259
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)	15. SECURITY CLASS. (of this report): UNCLASSIFIED
	15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Airworthiness and Flight Characteristics Evaluation
Compressibility Effects
Handling Qualities
Performance
Prime Item Development Specification
Vibration Characteristics
UH-60A (Black Hawk) Helicopter

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The United States Army Aviation Engineering Flight Activity conducted an Airworthiness and Flight Characteristics Evaluation of the third and fourth production UH-60A (Black Hawk) helicopters from 27 October 1979 through 9 October 1980. Performance, handling qualities, and vibration characteristics were evaluated to provide data for the operator's manual and to determine compliance with the applicable paragraphs of the Prime Item Development Specification. The UH-60A was tested at Edwards Air Force Base, California (elevation 2302 feet) and at alternate test site elevations of 488, 4120, and 9980 feet. The UH-60A was also tested in St. Paul, Minnesota (elevation 841 feet), to evaluate compressibility effects on level flight performance. A total of 129 flights were conducted on two test aircraft (US Army serial numbers 77-22716 and 77-22717)

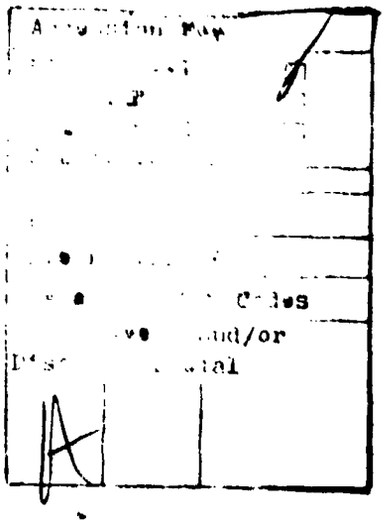
cont A

DD FORM 1 JAN 73 1473 EDITION OF 1 NOV 65 IS OBSOLETE

UNCLASSIFIED
SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

for a total of 86.1 productive flight hours. The UH-60A met 5 of the 7 performance commitments evaluated with reference to the Prime Item Development Specification. The performance of the UH-60A was better than the YUH-60A because of the lower primary mission gross weight of the aircraft, the reduced power required, and the increased power available. Due to the priorities set by United States Army Aviation Research and Development Command the handling qualities portion of the Airworthiness and Flight Characteristics Evaluation was extremely limited. The excellent engine torque matching greatly reduced pilot work load in the area of power management. When maneuvering, the limited load factor envelope at heavy gross weight was easily exceeded. The vibrations were found to be quite high in several areas and were considered to be excessive for a new generation helicopter. A total of 7 handling qualities-related deficiencies were identified (four of which were noted in previous evaluations) as follows: (1) the nose down pitching moment experienced during takeoff and climbout; (2) the inability of the pilot to control rotor speed in the electrical control unit lockout mode with the 4046T52G01 hydromechanical unit; (3) the unreliability of the automatic flight control system heading hold; (4) the design of the idle detent cam on the engine control quadrant which allows inadvertent engine shutdowns; (5) the inability to achieve full aft longitudinal cyclic control and the restriction of lateral cyclic control; (6) the uncommanded directional control input in trimmed forward flight; and (7) the erroneous information displayed by the horizontal situation indicator when the heading lockout submode of the command instrument system is engaged during an instrument landing system approach. A total of 64 shortcomings were identified, of which 38 were noted in previous evaluations.

DTIC COPY INSPECTED 3



DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
4300 GOODFELLOW BOULEVARD, ST. LOUIS, MO 63120

DRDAV-D

SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 77-17, Airworthiness and Flight Characteristics Evaluation, UH-60A (BLACK HAWK) Helicopter

SEE DISTRIBUTION

1. The purpose of this letter is to establish the Directorate for Development and Qualification's position on the subject report. The objectives of this Airworthiness and Flight Characteristics (A&FC) test were to obtain helicopter performance and handling qualities data for the UH-60A Operator's Manual and to determine compliance with the UH-60A Prime Item Development Specification (PIDS) for the production BLACK HAWK.

2. This Directorate agrees with the report conclusions and recommendations, with the exceptions identified herein. Dispositions of redesigned subsystems/components affecting the conclusion are also identified. Conclusions and recommendations are discussed by paragraph as indicated.

a. Paragraph 159b. The baseline handling qualities test conducted on the YUH-60A S/N 73-21650 by the contractor is reported in Sikorsky Aircraft report SER 70415 Volume 2, revision 2, 21 March 1980. This aircraft underwent the most extensive handling qualities optimization program ever performed for a US Army helicopter and these tests are well documented. The A&FC testing conducted by the US Army Aviation Engineering Flight Activity (USAAEFA) was adequate to address open issues not covered by the contractor. Such testing was minimal due to the minor differences between the contractor Maturity Phase prototype helicopter and the UH-60A used by USAAEFA for this A&FC. Additional handling qualities testing suggested by paragraph 30 of this report is not warranted.

b. Paragraph 161a. The nose down pitching moment deficiency resulted from large airspeed position errors which provided erroneous stabilator inputs. This becomes more serious as gross weight is increased. Developmental tests have been conducted which identified solutions to the problem and required modification to the existing airspeed system. Modification resulted in reorientation of the pitot static probes, increased stabilator electronic damping, pneumatic damping to the airspeed indicators, and reduction of stabilator trailing edge down positioning with collective. These modifications significantly reduce the nose down pitching moment and will be incorporated in the aircraft before fleet-wide use of the aircraft at increased gross weight is authorized.

c. Paragraph 161b. The engine ECU lockout with the 404T52G01 HMU deficiency has been corrected by changing out this unit in the field with the 404T52G04 unit.

d. Paragraph 161c. The unreliable AFCS heading hold and yaw excursion deficiency was corrected by incorporation of the -104 AFCS computer in the 71st and subsequent production UH-60A's. The earlier UH-60A's will be retrofitted with the -104 computers.

e. Paragraph 161d. The idle detent design deficiency which allows inadvertent engine shutdowns is being evaluated for cost impact. A redesign of the idle detent cam on the engine control quadrant is considered appropriate.

f. Paragraph 161e. The deficiency associated with the limited full aft longitudinal cyclic control and lateral cyclic control is being evaluated. The incorporation of a stop in the seat rail to limit forward seat travel is being considered.

g. Paragraph 161f. The uncommanded directional control input deficiency was corrected with the incorporation of the -104 AFCS computer and the vendor Acceptance Test Procedure (ATP) change for the roll and yaw trim actuators.

h. Paragraph 161g. As discussed in paragraph 97, and not 79 as stated in the report, erroneous information will be displayed on the HSI when missed approach instructions are set during an ILS approach. This stated deficiency represents incorrect procedure. The correct procedure, which is being taught at Ft. Rucker, is to have the pilot executing the approach set up his HSI information for the ILS approach and leave it there for the entire approach. The copilot (i.e., the individual not executing the approach) may place missed approach information on his HSI without affecting the ILS data on the pilot's HSI. This procedure could be reversed depending upon who is executing the approach. Therefore, the pilot executing the approach cannot set missed approach instructions until the actual missed approach point is reached. This procedure will be discussed at the next UH-60A Operator's Manual review to determine what information needs to be incorporated into the manual to preclude the individual executing the approach from setting missed approach instructions during an ILS approach.

i. Paragraph 162d. The large lateral control jump is not considered to be a shortcoming because it is strictly the result of pilot technique. The appropriate technique is for the pilot to engage the trim release after the stick forces are neutralized and not before.

j. Paragraph 162e. The lack of an ON/OFF switch on the aircraft's intercom system is not considered a shortcoming because the system is GFP. This item is more appropriately characterized as a suggested improvement.

k. Paragraph 162f. The lack of adequate aft seat adjustment is not considered a shortcoming since it meets the PIDS requirements. Additionally,

any further aft adjustment would allow the seat to intrude into the gunner's volume. Significant structural changes would be required to allow additional aft seat adjustment, therefore even as a suggested improvement, it does not appear to be a practical change.

l. Paragraph 162g. The poor engine droop characteristics certainly warrant consideration for improvement. The US Navy is funding the development of a collective anticipator for the SH-60B LAMPS which would improve the droop characteristic. This improvement could be available for future Army efforts if justified.

m. Paragraph 162h. The trim failure as stated is a shortcoming. However, this problem was re-evaluated on other production aircraft and could not be reproduced. Neither has the problem been reported from the field and therefore is considered an isolated case with no planned further action.

n. Paragraph 162i. The shortcoming associated with excessive vibrations during certain maneuvers has been corrected with the incorporation of redesigned vibration absorbers in the 240th and subsequent production aircraft. It is to be noted that the test aircraft contained the installation of a complete test instrumentation package which may have contributed to the excessive vibrations during the stated maneuvers.

o. Paragraph 162j. The pitch oscillation was investigated by Sikorsky Aircraft. The modifications to the airspeed system discussed in paragraph 2b above significantly reduced the pitch oscillations. No further action is considered necessary.

p. Paragraph 162k. The self excited divergent mechanical instability is not considered a shortcoming. It is difficult to excite this instability and it is easily controlled by the pilot when it occurs. The phenomena and pilot techniques are included in the Operator's Manual.

q. Paragraph 162l. The aircraft pitch oscillation is not considered a shortcoming. It is related to pilot technique and easily controlled. A NOTE identifying the pitch oscillation will be included in the Operator's Manual as follows: "The use of excessive collective pitch during taxi, especially at light gross weights, can cause the tail wheel to bounce."

r. Paragraph 162m. The lateral shuffle is not considered a shortcoming with respect to landing gear side loads as identified in the report. Per Sikorsky Aircraft report SER 70526 there is adequate tail wheel strength when landing from a hover with the lateral shuffle present.

s. Paragraph 162p. The shortcoming associated with the inability to use the No. 1 FM radio due to EMI problems has been eliminated by the incorporation of pin filter connectors into the stabilator amplifier. These connectors were incorporated in line on the 80th and subsequent UH-60A's and the stabilator

amplifiers in the 79th and earlier UH-60A's have been modified so that a restriction against the use of the No. 1 FM radio no longer applies.

t. Paragraph 162q. The lack of a mechanical forward cyclic control stop was corrected by incorporating a control stop in the 226th production helicopter and subsequent. Earlier helicopters will be retrofitted when product improvement funding is available.

u. Paragraph 162r. The limited load factor envelope is not considered an overall shortcoming. The conservative envelope used applied to the A&FC test per the AVRADCOM Airworthiness Release. The PIDS V-n diagram, which is greater than that released to AEFA, is applicable and the UH-60A has adequate structural margin to these limits. Therefore, a "g" meter is not required for operational aircraft.

v. Paragraph 162s. The shortcoming associated with the activation of the heading lockout submode by movement of the course set knob has been corrected by implementing proper operating procedures. VOR course changes to a new radial or identification of VOR intersections may be accomplished without going into the station passage mode (lockout submode) using this procedures now contained in paragraph 3-122 of the UH-60A Operator's Manual.

w. Paragraph 162t. The lack of an engine control quadrant lighting system is not considered a shortcoming since there is no PIDS requirement. This could be characterized as a suggested improvement.

x. Paragraph 162u. The lack of night lighting for the circuit breaker panels is not considered a shortcoming since there is no PIDS requirement. This could be characterized as a suggested improvement.

y. Paragraph 162w. The design of the IR suppressor is not considered a shortcoming. The indicated IR suppressor problem was related to an early IR suppressor material problem and not related to a production configuration. The report description is not totally correct as stated as it indicates fuselage skin damage in lieu of the actual IR suppressor skin problem. Further testing of the IR suppressor with high engine power has not shown any inherent problem with the installation.

z. Paragraph 162x. The shortcoming associated with partial obscuration of the warning lights has been corrected by a glare shield cut out on production aircraft.

aa. Paragraph 162y. The shortcoming associated with the erratic airspeed indications during aircraft accelerations below 60 KIAS will be corrected by the modifications to the airspeed system discussed in paragraph 2b above.

bb. Paragraph 162z. The shortcoming associated with the excessive frequency of illumination of chip lights has been corrected by the incorporation

of an improved, increased gap, chip detector on the 201st and subsequent production helicopters.

cc. Paragraph 162ff. The shortcoming associated with the lack of upper stops on the avionics compartment door was corrected by the incorporation of positive upper stops on the 162nd and subsequent production helicopters.

dd. Paragraph 162gg. The inability to easily determine the security of the APU compartment access door is not considered a shortcoming. The security of the APU compartment door can be determined by an adequate walkaround inspection.

ee. Paragraph 162ii. The shortcoming associated with the random latching of the AFCS computer maintenance latches was corrected by incorporation of the -104 computer in the 71st and subsequent production aircraft.

ff. Paragraph 162jj. The uncommanded engagements of the CIS modes were corrected through a maintenance action, i.e., replacement of the faulty CIS processor. This was not a design problem, therefore no other action is appropriate.

gg. Paragraph 162mm. The inability of the HSI of the CIS to display ADF and VOR navigational information simultaneously is not considered a shortcoming since the specification requirements have been met. This could be considered a suggested improvement.

hh. Paragraph 162pp. The inability of the pilot to utilize the VSI (roll bar) of the CIS during holding patterns at a VOR is not considered a shortcoming since the specification requirements have been met. This could be considered a suggested improvement.

ii. Paragraph 162rr. The lack of a copilot CIS mode select advisory panel is not considered a shortcoming since the specification requirements have been met. This could be considered a suggested improvement.

jj. Paragraph 162ss. The jarring pitch response to a cyclic release following a longitudinal cyclic displacement against trim is not considered a shortcoming. The jarring pitch response was the result of a test technique to evaluate dynamic response and stability. It does not represent an operational technique. This is clearly borne out by paragraph 47 which identifies the gust response of the UH-60A as an enhancing characteristic.

kk. Paragraph 162tt. The internal separations of the designated cabin-top step areas is not considered a shortcoming. Inspection of aircraft and review of structural analysis has resulted in a determination that the design is adequate. Field experience has not indicated this is a problem.

ll. Paragraph 162cc. The objectionably high cockpit and cabin noise levels are not considered a shortcoming. Although the absolute noise levels are high, the specification requirements for cockpit and cabin noise have been met. Further reductions in noise levels could be considered a suggested improvement.

DRDAV-D
SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 77-17, Airworthiness and Flight Characteristics Evaluation, UH-60A (BLACK HAWK) Helicopter

mm. Paragraph 162ggg. The shortcoming associated with the apparent low oil pressure as indicated by oil pressure gages and caution lights at low gas generator speeds has been corrected by the modification of both engine oil pressure scales on the Central Display Unit (CDU) so that the last uppermost red colored segment at the low range of the scale corresponds to 20 psi instead of 25 psi. This change has been incorporated on the 298th and subsequent production helicopters.

nn. Paragraph 162hhh. The use of multiple secondary lights/cockpit floodlight controls is not considered a shortcoming since the requirements of the UH-60A PIDS are met.

oo. Paragraph 162iii. The shortcoming associated with the failure of the external power source to immediately assume the aircraft electrical load upon APU shutdown has been corrected by an operational technique to be incorporated into the UH-60A Operator's Manual.

pp. Paragraph 162jjj. The lack of the proper alignment of the pilot and copilot slip indicators is not considered a shortcoming but rather faulty components in this particular BLACK HAWK tested. Other aircraft have proper alignment.

qq. Paragraph 162kkk. The various anomalies which occurred in the AFCS and CIS subsystem were peculiar to the aircraft being tested. These anomalies have not been reported on fielded aircraft.

rr. Paragraph 162lll. The shortcoming associated with the quick deterioration of the oleo strut splash guards has been corrected by removing the splash guards. This does not cause any detrimental effects to the oleo struts.

ss. Paragraph 165b, c, and d. Corrections to deficiencies and shortcomings are addressed in the preceding paragraphs. Additionally, corrections to those deficiencies and shortcomings not considered necessary are also addressed.

tt. Paragraph 165e. Consideration has been given to evaluating the handling qualities on a production UH-60A. However, based on the adequacy of testing conducted by AEFA and the contractor per paragraph 2a above, additional handling qualities testing was not considered warranted.

uu. Paragraph 165f. The recommended WARNING relative to slope operations will not be included in Chapter 8 of the UH-60A Operator's Manual. This type of information is common sense procedure and specific slope landing operations are adequately covered in the Operator's Manual. The inclusion of too much information in the Operator's Manual detracts from the truely pertinent data essential for safe and efficient flight operations.

vv. Paragraph 165g. The WARNING relative to the use of the radar altimeter settings for level-off commands will not be included in the UH-60A Operator's Manual. Paragraph 3-127, Level-Off Mode of the Operator's Manual includes this

information by stating the following: "During ILS or VOR approaches, the barometric altimeter must be used to determine arrival at the minimum altitude. Radar altimeter setting shall not be used for level-off commands in the VOR NAV/ILS modes because variations in terrain cause erroneous altitude indications."

ww. Paragraph 165h. The CAUTION relative to the lateral shuffle will not be incorporated since the landing gear strength is adequate for such a condition as discussed in paragraph 2r above and the additional information required to be memorized by the pilot is unwarranted.

xx. Paragraph 165i. The CAUTION relative to APU compartment access door security will not be incorporated since this function is properly executed during a walkaround inspection.

yy. Paragraph 165j. The definition of ground resonance to include limitations and corrective actions is adequately covered in the UH-60A Operator's Manual; therefore the recommended CAUTION relative to the definition of conditions which could result in ground resonance will not be included.

zz. Paragraph 165k. A brightly colored strip will not be incorporated as such an item will compromise the camouflage scheme of the aircraft. A proper walkaround inspection should preclude the necessity for such an item.

aaa. Paragraph 165l. A NOTE relative to rapid collective reductions resulting in a transient main rotor speed droop of approximately 2% will not be incorporated since specification requirements have been met and it is not considered essential that this condition be highlighted.

bbb. Paragraph 165m. The limits currently published in paragraph 5-31.2 of the UH-60A Operator's Manual are not being changed. Monitoring rotor clearance during slope landings is part of the normal operating procedure. Additionally, the Operator's Manual currently limits the aircraft to 6° nose downslope vice the 10° nose downslope recommended.

ccc. Paragraph 165r. Further evaluating the significance of heat damage during high power hover with the IR subsystem installed is not planned per discussions in paragraph 2y above.

ddd. Paragraph 165s. All engine changes have been assessed with no further effort required except as required due to future engine changes.

eee. Paragraph 165u. The contractor conducted runaway trim tests on a modified prototype aircraft/production control system configuration prior to the AEFA tests. Contractor tests indicated no problem with runaway trims about any axis, therefore no further tests are planned.

fff. Paragraph 165v. Although we believe adequate testing has been conducted by both AEFA and the contractor to define control margins and the

current UH-60A operating limits apply for any gross weight or center of gravity, additional slope landing evaluations at extreme cg locations will be added to the next convenient UH-60A AEFA test.

ggg. Paragraph 165dd. A "g" meter will not be installed in the UH-60A for the reasons discussed in paragraph 2u above.

hhh. Paragraph 165gg. A TGT limiter which is automatically disabled for single engine operation will be investigated relative to a system like the contingency power enable feature on the T700-GE-701 engine. This feature sets a higher TGT limit when one engine is off-line.

3. The UH-60A helicopter is considered qualified based on all the testing accomplished by AEFA and the contractor. It has undergone the most extensive qualification program of any helicopter in the history of the industry. At this time, additional flight testing for the basic helicopter configuration is not warranted. Any expansion of the gross weight or center of gravity limits or incorporation of additional subsystems will require further flight testing to substantiate airworthiness.

4. Although the formal distribution of this report is occurring substantially after the test, it is important to note that all organizations within the Army were provided advance copies of the report so that appropriate actions could be taken. In addition to these internal efforts, it is most significant that the actual flight performance data contained herein was transmitted to the contractor 15 Jun 82 for his use in preparation of an updated Chapter 7 for the Operator's Manual. The importance of the test and actions taken as a result of the test are clearly illustrated by the large number of identified problems for which solutions are now available as noted throughout this position letter.

FOR THE COMMANDER:

CHARLES C. CRAWFORD, JR.
Director of Development
and Qualification

TABLE OF CONTENTS

Page

INTRODUCTION

Background 1
Test Objectives 1
Description 1
Test Scope 3
Test Methodology 3

RESULTS AND DISCUSSION

General 6
Performance 6
General 6
Hover Performance 6
Takeoff Performance 7
Vertical Climb Performance 8
Forward Flight Climb Performance 8
Level Flight Performance 8
Autorotational Descent Performance 11
Handling Qualities 12
General 12
Control System Characteristics 12
Control Positions in Trimmed Forward Flight 19
Normal Utility Configuration (Level Flight) 19
Normal Utility Configuration Cargo Doors and Gunner Windows Fully Open 19
Infrared Reduction Subsystem, XM-130 and, AN/ALQ-144 Configuration 20
Normal Utility Configuration (Climbs and Autorotations) . . . 20
Static Longitudinal Stability 20
Maneuvering Stability 21
Dynamic Stability (Gust Response) 21
Controllability 21
Ground Handling Characteristics 23
Takeoff Characteristics 25
Slope Landing Evaluation 30
Low Speed Flight Characteristics 32
Power Management 33
Instrument Flight Characteristics 35
In-flight Engine Starts 39
Aircraft System Failures 41
Automatic Flight Control System Failure 41
Pitch Bias Actuator Failures 41
Vibration Characteristics 41
Human Factors 43
General 43
Cockpit Evaluation 43
Night Evaluation 47
Noise Evaluation 47
Reliability, Availability and Maintainability 48
Airspeed Calibration 51
Miscellaneous 52

CONCLUSIONS

General 53
Enhancing Characteristics 53
Deficiencies 53
Shortcomings 54
Specification Compliance 57

RECOMMENDATIONS 59

APPENDIXES

A. References 63
B. Aircraft Description 65
C. Instrumentation 98
D. Test Techniques and Data Analysis Methods 105
E. Test Data 122
F. Glossary 249
G. Equipment Performance Reports 251

DISTRIBUTION

INTRODUCTION

BACKGROUND

1. The United States Army requires an improved operational capability in its utility transport aircraft to satisfy the demand for increased performance and survivability in the mid-intensity combat environment. The utility tactical transport aircraft system was developed in response to this requirement and will augment the utility helicopters in the current Army inventory. Its primary and secondary mission will include both all-weather and night capability of transporting a variety of internal loads and a capability of transporting external loads under visual meteorological conditions. More specific mission requirements are contained in the Material Need Document (ref 1, app A).

2. On 23 December 1976, the United States Army Aviation System Command redesignated the United States Army Aviation Research and Development Command (AVRADCOM), awarded a contract to Sikorsky Aircraft Division of United Technologies Corporation for follow-on maturity phase/verification testing of the UH-60A helicopter. In August 1977, the United States Army Aviation Engineering Flight Activity (USAAEFA) was tasked by AVRADCOM (ref 2, app A) to plan, conduct, and report on the airworthiness and flight characteristics (A&FC) evaluation of the UH-60A helicopter, also designated Black Hawk.

TEST OBJECTIVES

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

a. To obtain sufficient performance data to establish a basis for the performance information in the operator's manual

b. To obtain sufficient handling qualities data for inclusion in the operator's manual

c. To determine compliance with the applicable paragraphs of the prime item development specification (PIDS) (ref 3, app A).

DESCRIPTION

4. The test helicopters, UH-60A US Army S/N 77-22716 and S/N 77-22717 (photo 1), are the third and fourth production Black Hawks. The UH-60A is a twin engine, single main rotor configured helicopter with nonretractable wheel-type landing gear. A movable horizontal stabilator is located on the lower portion of the tail rotor pylon. The main and tail rotor 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. The tail rotor shaft is canted 20 degrees upward from the horizontal. Primary mission gross weight is 16,260 pounds and maximum alternate gross weight is 20,250 pounds. The UH-60A is powered by two General Electric (GE) T700-GE-700 turboshaft engines having an installed power available (30 minute limit) of 1553 shaft horsepower (shp) (power turbine speed of 20,900 revolutions per minute (rpm)) each at sea level, standard-day static conditions. Installed dual-engine power is transmission limited to 2828 shp. The T700-GE-700 engine incorporates a history recorder, automatic

Photo 1. UH-60A (Black Hawk)

turbine gas temperature (TGT) limiter, power turbine (N_p) speed limiter, gas turbine (N_g) speed limiter, automatic torque-matching capability and, various diagnostic systems. The aircraft also has an automatic flight control system (AFCS) and a command instrument system (CIS). A more detailed description of the UH-60A is included in appendix B and additional descriptions can be found in the operator's manual (ref 4, app A).

TEST SCOPE

5. The major portion of flight testing was conducted at Edwards Air Force Base, California (2302 feet). Performance flight testing was also conducted at St. Paul, Minnesota (841 feet), Bakersfield (488 feet), Bishop (4120 feet), and Coyote Flats (9980 feet), California. A total of 129 flights were conducted between 27 October 1979 and 9 October 1980 on two test aircraft (US Army serial numbers 77-22716 and 77-22717) for a total of 86.1 productive flight hours. USAAEFA calibrated and maintained all the test instrumentation and performed all required maintenance on both helicopters. Flight restrictions and operating limitations observed during the A&FC are contained in the operator's manual (ref 4, app A) and the airworthiness release (refs 5 and 6). Testing was conducted in accordance with the test plan (ref 7) at the conditions shown in tables 1 and 2, which were based on the commitments of the PIDS. All previous UH-60A problems identified by USAAEFA excluding icing reports and problems listed in paragraph 157 have been re-evaluated.

TEST METHODOLOGY

6. A detailed listing of the test instrumentation is contained in appendix C. Established flight test techniques and data reduction procedures were used (refs 8 and 9, app A), and are described in appendix D. A Handling Qualities Rating Scale (HQRS) (fig. 4, app D) and a Vibrations Rating Scale (VRS) (fig. 5) were used to augment pilot comments relative to aircraft handling qualities and vibrations. The flight test data were obtained from test instrumentation displayed on the instrument panel and recorded on magnetic tape installed in the aircraft. Real time telemetry monitoring of selected critical data parameters was used during certain tests.

Table 1. Performance Test Conditions[1]

Type of Test	Gross Weight (lb)	Center of Gravity Long. (FS)	Center of Gravity Lat. (BL)	Density Altitude (ft)	Trim Airspeed (kt)
Hover	14,030 to 21, 400[2]	347.2 (fwd)	0.0 (mid)	2100 to 11,480	0
Vertical Climb	16,000 to 19,300	347.2 (fwd)	0.0 (mid)	2540 to 4700	0
Forward Flight Climb (K_p[3] and K_w[4])	14,540 to 19,500	347.3 (fwd)	-0.2 (lt)	2500 to 15,000	70 to 75 KCAS[5] (climb speed schedule)
Level Flight	15,500 to 19,980 17,160 16,460[7] 16,880 to 19,800[8]	347.1 (fwd) 358.8 (aft) 347.2 (fwd) 347.1 (fwd)	0.0 (mid) 0.0 (mid) 0.0 (mid) 0.0 (mid)	2060 to 14,740 4740 to 11,060 5960 to 12,620 8780 to 12,460	48 to 166 KTAS[6] 59 to 165 KTAS 58 to 157 KTAS 66 to 154 KTAS
Autorotation	16,200 to 19,760	347.4 (fwd)	-0.2 (lt)	5120 to 5460	43 to 130 KCAS

NOTES:

[1] Tests conducted in the normal utility configuration with all doors closed unless otherwise noted.
[2] Aircraft weight plus cable tension
[3] K_p = Power correction factor
[4] K_w = Weight correction factor
[5] KCAS = Knots calibrated airspeed
[6] KTAS = Knots true airspeed
[7] Normal utility configuration with cargo doors and gunner windows fully open.
[8] Aircraft configured with an infrared suppressor, XM-130, and AN/ALQ-144

Table 2. Handling Qualities General Test Conditions[1]

Type of Test	Gross Weight (lb)	Center of Gravity		Density Altitude (ft)	Trim Calibrated Airspeed (kt)
		Long. (FS)	Lat. (BL)		
Control Positions in Trimmed Forward Flight[2]	16,120 to 19,760	347.0 (fwd)	0.0 (mid)	6060 to 9420	43 to 152
	17,160	358.8 (aft)	0.0 (mid)	4560	57 to 153
Static Longitudinal Stability	15,180 to 16,640	360.2 (aft)	0.1 (rt)	7620 to 7980	88, 123 and 153
Maneuvering Stability	16,740	359.0 (aft)	0.2 (rt)	6520	118
	15,400	358.5 (aft)	0.2 (rt)	8400	131 (PBA locked)[3]
Controllability	18,680 to 19,920	359.4 (aft)	0.2 (rt)	6420 to 6780	90 and 140
Takeoff Characteristics	16,220 to 21,760	347.3 (fwd) to 352.2 (mid)	0.0 (mid)	3420 to 4560	zero to 120
Slope Landings	16,460	352.7 (mid)	0.0 (mid)	4160 to 4320	zero
Low Speed Flight Characteristics	16,840	347.0 (fwd)	0.2 (lt)	9860	zero to 47 KTAS[4]
	16,360	346.6 (fwd)	1.3 (lt)	2020	zero to 45 KTAS (IR config.)
In flight Engine Starts	13,580 to 14,180	359.5 (aft) to 364.3 (aft)	0.1 (rt)	13,960 to 20,020	30 to 92
Vibrations	13,360 to 19,780	347.1 (fwd)	0.0 (mid)	6180 to 14,740	42 to 153 Level Flight
	16,740 to 19,720	362.0 (aft)	0.2 (rt)	6520	118 lt & rt steady turns

NOTES

[1] Normal utility configuration, automatic flight control system ON and 100% main rotor speed
[2] Both normal utility and IR configurations (IR - infrared suppressor, XM-130 and AN/ALQ-144 systems installed)
[3] PBA locked - pitch bias actuator centered and locked
[4] KTAS - knots true airspeed

RESULTS AND DISCUSSION

GENERAL

7. The performance and handling qualities of the UH-60A were evaluated under a variety of operating conditions at test sites from near sea level (488 feet) to 9980 feet. The UH-60A met 5 of the 7 performance requirements evaluated with reference to the PIDS. The performance of the UH-60A improved over that of the YUH-60A because of the lower primary mission gross weight of the aircraft, the lower power required, and the increased power available. The UH-60A exhibited three features which enhance accomplishment of the Black Hawk mission, one of which was the excellent engine torque matching which greatly reduced pilot work load in the area of power management. When maneuvering, the limited load factor envelope at heavy gross weight was easily exceeded. The vibrations were found to be quite high in several areas and were considered to be excessive for a new generation helicopter. A total of seven handling qualities-related deficiencies were identified (four of which were noted in previous evaluations) as follows: (1) the nose down pitching moment experienced during takeoff and climbout; (2) the inability of the pilot to control rotor speed in the electrical control unit lockout mode with the 4046T52G01 hydromechanical unit; (3) the unreliability of the AFCS heading hold; (4) the design of the idle detent cam on the engine control quadrant which allows inadvertent engine shutdowns; (5) the inability to achieve full aft longitudinal cyclic control and the restriction of lateral cyclic control; (6) the uncommanded directional control input in trimmed forward flight; and (7) the erroneous information displayed by the horizontal situation indicator when the heading lockout submode is engaged during an instrument landing system approach. A total of 64 shortcomings were identified, of which 38 were noted in previous evaluations. All previous problems stated by USAAEFA excluding icing reports and problems listed in paragraph 157 have been re-evaluated.

PERFORMANCE

General

8. Performance flight testing was conducted at test site elevations of 488, 2302, 4120 and 9980 feet. Testing was also conducted at St. Paul, Minnesota to evaluate mach number (compressibility) effects on performance during level flight. Performance evaluations included tethered hover, vertical climb, forward flight climb, level flight, and autorotational descent. Power available and fuel flow was based on data received from AVRADCOM (ref 10, app A). The data is an average of the left and right engines derived from the GE deck number 80024, dated 26 February 1981 using installation losses determined by AVRADCOM. The UH-60A met 5 of the 7 performance requirements evaluated with reference to the PIDS. The performance of the UH-60A improved over that of the YUH-60A because of the lower primary mission gross weight of the aircraft, the lower power required, and the increased power available.

Hover Performance

9. Hovering tests were conducted utilizing the tethered hover method at the conditions of table 1. The 2-foot main wheel height in ground effect (IGE) and 100 foot main wheel height out of ground effect (OGE) tests were conducted at the 2302, 4120, and 9980 foot test sites. The 5-foot main wheel height IGE tests were conducted at only the 2302 foot and 4120 foot test sites. A cable tensiometer was

used to measure total thrust less gross weight. Variations in the coefficient of thrust (C_T) were attained by varying rotor speed from 95 to 103 percent (245 to 265 rpm) and tension in the cable. The nondimensional test results appeared to be a function of density altitude. An increase in density altitude requires a corresponding increase in power coefficient required to maintain a constant hover height at a specific C_T. Tests were also conducted OGE with an infrared (IR) suppressor, XM-130 chaff dispenser, and an AN/ALQ-144 IR jammer installed on the aircraft (IR configuration). Hover test results are presented in figures 1 through 10, appendix E.

10. The standard day OGE hover ceiling at the primary mission gross weight of 16,260 pounds was 11,200 feet using intermediate rated power (IRP) available obtained from AVRADCOM (para 5, app D). At 4000 feet, pressure altitude (H_p) on a 35° C day, the OGE hover maximum gross weight was 17,721 pounds with IRP. There was approximately 4 percent increase in power required to hover OGE with the aircraft in the IR configuration.

11. At the hover performance guarantee conditions of 95 percent IRP at 4700 feet H_p, on a 35° C day the OGE hover maximum gross weight was 16,570 pounds. This differs from the 16,364 pounds previously reported in the Production Validation Test-Government (PVT-G), Performance Guarantees UH-60A Black Hawk Helicopter, Final Report (ref 11, app A) by 206 pounds of increased hover capability. Some of the increase is due to an increase in power available of 9 shp (ref 10, app A). There are other unexplained discrepencies between the PVT-G and A&FC hover results. However, some differences may be attributed to the C_T range, rotor speed, wheel heights, and site elevations. Also, the data analysis technique was different for the A&FC which was based on the best curve fit through the large C_T range (relatable to theory) as opposed to the statistical curve fit through the small C_T range used during the PVT-G.

Takeoff Performance

12. Quantitative data were not successfully obtained during this test program to evaluate the takeoff performance of the UH-60A helicopter. The PIDS, paragraph 3.2.1.1.1.3.d, requires the UH-60A helicopter to be able to accomplish a single engine takeoff from a 2-foot hover at 4000 feet H_p on a 35° C day at a gross weight of 12,794 pounds. This gross weight is defined as the primary mission (troop assault) gross weight (16,260 pounds) minus the sum of 11 troops (2640 pounds) and 40 percent of the mission fuel (826 pounds).

13. Several takeoff attempts from a 2-foot hover with simulated zero excess power resulted in ground contact just after the initial pitch attitude change during the takeoff attempt. The power condition was simulated by maintaining the collective control fixed throughout the maneuver at the position required to hover the aircraft. The only technique used for this evaluation was an attempted level acceleration from a hover at a 2-foot main wheel height. This same result is evident from the Government Competitive Test (GCT) data (ref 12, app A). Takeoff performance data were obtained using the same technique but from a 5-foot hover height. It was reported that the helicopter lost up to 3 feet of altitude during the initial part of the takeoff when a zero excess power condition was simulated. This 3-foot loss and the recent flight test data indicate that the UH-60A helicopter is not capable of a zero excess thrust takeoff from a 2-foot hover height without ground contact.

14. At the atmospheric conditions required by the PIDS, using single engine IRP, the aircraft will hover at a 2-foot main wheel height at 12,956 pounds, 162 pounds heavier than the PIDS takeoff requirement. From the hover data obtained at various wheel heights, it was determined that the aircraft could hover at a main wheel height of approximately 3 feet at these same conditions. It is questionable if the UH-60A helicopter can accomplish a single engine takeoff at 4000 feet H_p on a 35° C day, at a gross weight of 12,794 pounds from a 2-foot hover without ground contact.

Vertical Climb Performance

15. Vertical climb tests (zero horizontal speed) were conducted at the conditions of table 1. Three series of vertical climbs were accomplished at a constant ratio of gross weight to pressure ratio (W/δ) and at a referred rotor speed ($N_R/\sqrt{\theta}$) of 253 rpm. The vertical climbs were initiated from an OGE hover using various constant collective control settings throughout the dual engine power range. The vertical rate of climb for a given power was defined as that portion of the climb after the aircraft had achieved a steady unaccelerated rate of climb condition. Vertical climb test results are presented in figures 11 and 12, appendix E.

16. At sea level, on a 35° C day, at the primary mission gross weight the maximum vertical rate of climb was 1783 feet per minute (ft/min) with 95 percent IRP. At a pressure altitude of 4000 feet, 35° C, 95 percent IRP and primary mission gross weight the maximum rate of climb was 590 ft/min. This performance exceeds the rate of climb of 450 ft/min of paragraph 3.2.1.1.1.1.A of the PIDS.

Forward Flight Climb Performance

17. Continuous climbs were conducted from near sea level to each respective climb service ceiling (altitude at which a 100 ft/min rate of climb is the maximum achievable) or 15,000 feet H_p to determine the power and weight correction factors (K_p and K_w). The climb tests were conducted at the conditions listed in table 1. The climb tests were flown at the airspeed schedule for best rate of climb as determined from the level flight performance. The power schedules used were percentages of MCP based on the test day power available. Test results are presented in figures 13 and 14, appendix E. The K_p was determined to be 0.76. The K_w was found to vary as a function of gross weight throughout the altitude range tested.

18. The single engine climb performance at the primary mission gross weight met the PIDS requirement in that the rate of climb at 5000 H_p and 35° C day conditions exceeded the 100 ft/min of paragraph 3.2.1.1.13 by 40 ft/min.

Level Flight Performance

19. 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 the operational envelope of the aircraft. Techniques used in obtaining and analyzing level flight performance data are described in detail in appendix D. The aircraft was flown in ball centered flight to obtain the data, and then converted to a zero sideslip trim condition for analysis. All performance data were corrected for estimated drag of external test instrumentation and instrumentation electrical load.

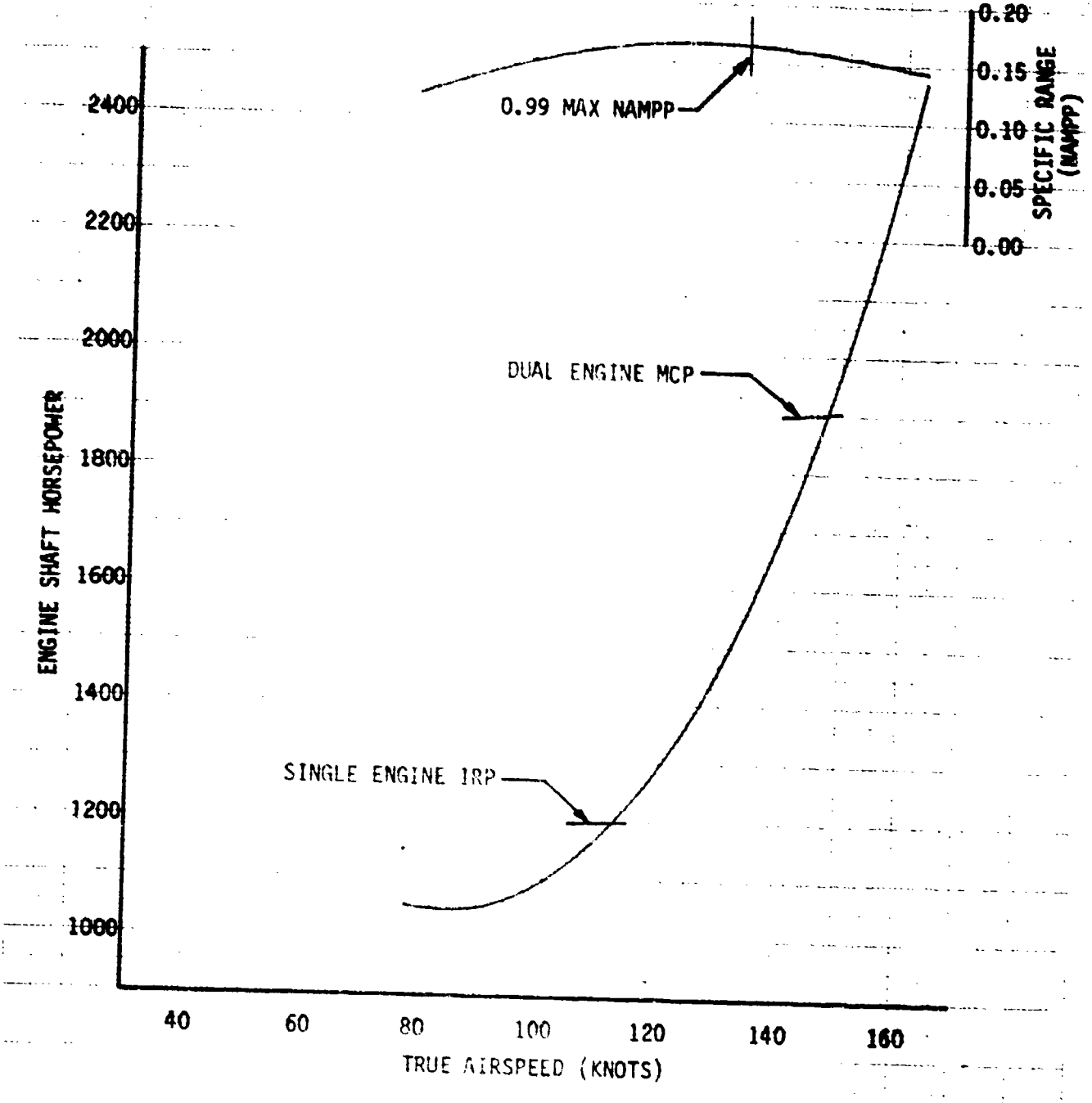
20. Nondimensional test results are presented in figures 15 through 42, appendix F. The data indicate that for $N_R/\sqrt{\theta}$ between 258 and 275 rpm (maximum tested) there is not any change in power required for similar advance ratios (μ) for C_T's through 80×10^{-4}. Beyond this point, power required increases with increasing $N_R/\sqrt{\theta}$ for either constant C_T or μ, attributable presumably to compressibility effects. Below $N_R/\sqrt{\theta}$ of 258, power required decreases for either constant C_T or μ (for μ values above those corresponding to minimum power required) due to undetermined causes. Inherent sideslip, presented in figures 43 through 45, was developed from resultant angle of sideslip associated with ball centered flight during level flight performance testing. The results are independent of rotor speed and applicable throughout the longitudinal center of gravity (cg) range and all aircraft configurations tested. The data indicate that as airspeed increases, inherent sideslip converges for all C_T's. Change in equivalent flat plate area (ΔF_e) as a function of sideslip angle is presented in figure 46. This curve was determined to be independent of airspeed, C_T, and $N_R/\sqrt{\theta}$. The data indicate that minimal F_e corresponds to a left sideslip angle of approximately 7 degrees. Coordinated flight between best endurance airspeed and maximum airspeed in level flight at intermediate power (V_H) normally occurs between 5 and 0 degrees left sideslip. Dimensional level flight test results are presented in figures 47 through 76. All dimensional test results and PIDS compliance calculations are presented for ball centered flight conditions as determined by the method in appendix D.

21. The primary mission of the UH-60A is defined in paragraph 3.2.1.1.1.1 of the PIDS, with performance provisions provided in paragraph 3.2.1.1.1, and stipulates operation in 4000 feet H_p and 35° C environment. Primary mission gross weight is defined by paragraph 3.2.2.1.5 of the PIDS to be 16,260 pounds, and was used as the basis for primary mission level flight performance computations. Computations to determine compliance with the PIDS involved extrapolation beyond the range of level flight performance data accumulated during the A&FC evaluation. Primary mission cruise speed of 145 knots true airspeed (KTAS) was exceeded by over 2 knots (fig. A) at maximum continuous power (MCP) and meets the requirement of paragraph 3.2.1.1.1 1.a of the PIDS. However, long range cruise speed based on 99 percent of the maximum nautical air miles per pound of fuel was 133 KTAS. Single engine V_H using IRP at primary mission conditions exceeded the level flight speed requirement of 105 KTAS by over 7 knots (fig. A) and meets the requirement of paragraph 3.2.1.1.1.3.a of the PIDS. However, at MCP on the remaining engine the UH-60A is unable to maintain level flight at any airspeed. The primary mission requires an endurance of 2.3 hours based on the following mission profile: 8 minutes ground operation at idle power; 20 minutes operation at MCP; 80 minutes cruise at 145 KTAS; 30 minutes fuel reserve at 145 KTAS. Endurance calculations provided for gross weight to be decreased for fuel burn-off, fuel specific weight to be based on specified temperatures, and fuel flow to be increased by 5 percent. The maximum endurance of the UH-60A was 2.57 hours, assuming the extra 16 minutes available were used during cruise at 145 KTAS, and exceeds the requirement of paragraph 3.2.1.1.1.1.c of the PIDS by 0.27 hours.

22. The PIDS also specifies an alternate endurance requirement of 2.3 hours for the same mission profile as defined in paragraph 20 at sea level, standard day conditions. Gross weight for this alternate mission is defined by paragraph 3.2.2.4.1.1 of the PIDS as the basic structural design weight of the aircraft and is 16,825 pounds. Maximum endurance of the UH-60A under these circumstances was 2.16 hours, allowing for 22 minutes fuel reserve instead of the required 30 minutes, and fails to meet the requirement of paragraph 3.2.1.1.1.2 of the PIDS.

FIGURE A
PRIMARY MISSION LEVEL FLIGHT PERFORMANCE

GROSS WEIGHT (LB)	LONGITUDINAL CG (FS)	PRESSURE ALTITUDE (FT)	AIR TEMPERATURE (° C)	ROTOR SPEED (RPM)	C_T X 10^4	CONFIGURATION
16,260	FWD	4000	35	258	71.24	NORM UTIL



23. Testing was accomplished at other than a forward cg and normal utility configuration in order to determine the comparative effects on level flight power required. Though small unexplainable inconsistencies of ΔF_e with C_T exist for aft cg and doors open configurations, the nominal ΔF_e value presented is considered to be representative for all C_T's. Changing the longitudinal cg approximately 12 inches changes ΔF_e by 2 ft^2. With the cargo doors and gunner windows fully open, the ΔF_e increased by 6 ft^2. This equates to an increase of approximately 70 shp to maintain limit airspeed (100 knots indicated airspeed (KIAS)) (fig. 72). Installing an infrared suppressor, XM-130 chaff dispenser, and an AN/ALQ-144 infrared jammer increased ΔF_e by 7.5 ft^2. This results in a reduction of airspeed of 6 knots at MCP when compared to the normal utility configuration (fig. 74).

Autorotational Descent Performance

24. Autorotational descent performance tests were conducted near the mission and alternate gross weight at the conditions listed in table 1. The tests were conducted by retarding the power control levers to the idle position and then stabilizing the aircraft on an airspeed and rotor speed. At the normal operating rotor speed of 100 percent (258 rpm), airspeed was varied to determine the airspeed for minimum rate of descent ($V_{min\ R/D}$). At the approximate airspeed for minimum rate of descent, rotor speed was varied to determine the effect on descent performance. Test results are presented in figures 77 and 78, appendix E.

25. The minimum rate of descent for both the mission and alternate gross weights was 2060 ft/min at an airspeed of 67 knots calibrated airspeed (KCAS). Airspeed can vary ±5 KCAS without increasing the rate of descent more than 1 percent. This 10 knot band is a desirable characteristic and allows the pilot to concentrate on other duties without significantly increasing the rate of descent. The airspeed for maximum glide distance was 105 KCAS and the rate of descent was 2540 ft/min. Minimal pilot effort was required to maintain the airspeed for minimum rate of descent or maximum glide distance (HQRS 3).

26. Field of view during steady state autorotation did not change significantly from that observed during level flight as only a 1 to 2 degree nose up pitch attitude change was required. Trim airspeed for determining the effects of rotor speed on rate of descent was 76 KCAS at the mission gross weight and 73 KCAS for the alternate gross weight. The rate of descent varied from 2020 ft/min (238 rpm) to 2380 ft/min (282 rpm) at a gross weight of 16,500 pounds. The rotor speed for minimum rate of descent was 238 rpm at 16,500 pounds and 245 at 19,760 pounds. This resulted in a minimum rate of descent of 2020 ft/min and 2065 ft/min respectively.

27. The UH-60A exhibited a tendency to underspeed the rotor system during rapid collective reductions with both engines operating. A 3-inch reduction of collective in 1 second resulted in a main rotor transient droop of 2 percent; however, the rotor speed recovered to 100 percent within 3 seconds of the initial droop. The following note should be placed in the operator's manual:

NOTE

Rapid collective reductions may result in a transient main rotor speed droop of approximately 2 percent.

28. Rotor speed control during entry into autorotational flight as well as during steady state autorotation required moderate pilot compensation to maintain the desired rotor speed (HQRS 4). The sensitivity of rotor speed to collective movement detracts from the pilot's ability to maintain visual contact outside the cockpit and is a shortcoming.

HANDLING QUALITIES

General

29. Stability and control, operational, and limited system tests were conducted to qualitatively and quantitatively evaluate the handling qualities of the UH-60A helicopter. Since AVRADCOM placed higher priority on additional tests and the sequence of testing, and all portions of the original test plan (ref 7) could not be completed within cost and schedule limitations, the handling qualities portion of the A&FC was limited. Previous tests were conducted at 13,000 pounds at an aft cg (FS 365) which showed that the longitudinal handling qualities were satisfactory (ref 13, app A). The excellent torque matching capability and aircraft gust response characteristics with the AFCS engaged greatly reduced pilot workload and are enhancing characteristics. In maneuvering the aircraft at heavy gross weights the limited load factor envelope was easily exceeded.

30. The handling qualities testing during the A&FC evaluation was extremely limited. As a result, there has been very little U.S. Army base line handling qualities data on the production UH-60A. At a minimum, consideration should be given to evaluating the following handling qualities on a current production UH-60A:

a. Static longitudinal stability in climbs, level, and descending flight

b. Static lateral-directional stability in climbs, level, and descending flight

c. High airspeed maneuvering stability

d. Short and long term dynamic stability in climbs, level, and descending flight

e. Controllability at a hover

f. Single hydraulic failures.

Control System Characteristics

31. The control system characteristics of the UH-60A were evaluated on the ground with external hydraulic and electrical power applied to the aircraft, engines and rotor static, and with all AFCS engaged. The results of the cyclic and collective control systems tests, obtained by on-board instrumentation and a hand-held force gage, are presented in figure 79 through 84, appendix E. The flight control system mechanical characteristics are summarized in tables 3 through 6. Control forces were measured at the center of the cyclic and collective grips. The control system characteristics were also qualitatively evaluated in-flight with all functions of the AFCS engaged. Other than the lateral control jump (para 33), the results of the ground evaluation qualitatively agreed with the in-flight results.

32. The UH-60A has varying degrees of mechanical and electrical control coupling which are described in figure 6, appendix B. A summary of cyclic control travel limits for selected directional and collective positions is presented in figure 79, appendix E. It should be noted that the cyclic control is limited by either flight control stops or limiters located in the mixing unit. Figures 80 and 81, and table 3 summarize the longitudinal control system characteristics. Longitudinal control exhibited positive control centering with a force versus displacement gradient of 0.9 lb/in. forward and 1.3 lb/in. aft without any significant discontinuities. The breakout force (plus friction) was 0.7 pounds forward and 0.4 pounds aft which failed to meet the requirements of paragraphs 10.3.2.1.2 and 10.3.2.1.1, respectively, of the PIDS in that the forward and aft breakout force (plus friction) were not symmetric and the aft force was 0.1 pounds below specification. However, the lack of symmetry and the low aft breakout force (plus friction) were not perceivable in flight and are acceptable. Longitudinal control jump was 0.1 inches both forward and aft after trim switch activation following a 20 percent control displacement against trim. The longitudinal control jump failed to meet the requirement of paragraph 10.3.3.1.1.1 of the PIDS which specifics control jump shall not occur. In flight the longitudinal control jump was considered to be insignificant by the pilots. The longitudinal control exhibited six overshoots when displaced forward from trim and released, and three overshoots when displaced aft from trim and released during the control dynamics test (10 percent control deflection). The same test was then conducted during level flight and the oscillations were not transmitted to the airframe. However, a jarring pitch response was observed and is discussed further in paragraph 52.

33. During the ground control system characteristics check, it was noted that the forward longitudinal cyclic stop (located at the base of the cyclic) had been removed by a production modification. With collective full up and pedals centered, the forward cyclic limiter was reached with 7 pounds of push force (longitudinal cyclic trimmed at 50 percent). Any additional push force was absorbed by deflection of the mechanical linkage between the cyclic control and the boost actuator. The lack of a positive forward control stop in the longitudinal control system creates a problem during preflight in that the pilots used excessive force when attempting to determine the forward longitudinal control travel. This excessive force was due to the ill-defined forward control limit. Additionally, the repeated use of excessive force may result in long term structural fatigue problems. With the control stops placed in the base of the cyclic control, the limits of travel are well defined by a firm and positive opposing force. Consideration should be given to reinstalling the forward longitudinal cyclic stick stop to preclude the long term excessive forces having a detrimental effect. The UH-60A failed to meet the requirements of MIL-F-18372, paragraph 3.1.1.2.2 (ref 14, app A) as referenced in the PIDS (para 3.7.6.1) in that the forward stop is not located as near the cockpit control as possible. The lack of a mechanical forward cyclic control stop in the longitudinal control system is a shortcoming previously reported in the Preliminary Airworthiness Evaluation (PAE) III, Apr 1979 (ref 15, app A).

34. The lateral control system characteristics are summarized in figures 82 and 83, appendix E and table 4. The breakout force (plus friction) was 1.9 pounds to the left and 1.5 pounds to the right. As in the longitudinal axis, the breakout force (plus friction) was not symmetric in the lateral axis and failed to meet the symmetry requirements of paragraph 10.3.2.1.2 of the PIDS. However, the lack of symmetry in lateral breakout force (plus friction) was not considered to be significant in flight.

Table 5. Longitudinal Control System

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Breakout force plus friction (lb)	0.7 (fwd) 0.4 (aft)	Par. 10.3.2.1.1 0.5 min, 2.6 max	Fwd, yes Aft, no
		Par. 10.3.2.1.2 Symmetrical within 10 pct	No
Control Centering	Positive	Par. 10.3.2.2 Positive	Yes
Force versus displacement gradient (lb/in.)	0.9 (fwd) 1.3 (aft)	Par. 10.3.2.2 0.4 min, 3.0 max	Yes
	No discontinuities	Par. 10.3.2.2.1 Positive without discontinuities	Yes
Slope of force gradient for first inch of travel from trim (lb/in.)	1.7 (fwd) 2.7 (aft)	Par. 10.3.2.2.1 Slope for the first inch must be equal to or greater than the slope for remaining control travel.	Yes
Control forces trimmable to zero	Yes	Par. 10.3.2.5 Required	Yes
Control forces maintained at zero	Yes	Par. 10.3.2.5 Required	Yes
Control system freeplay[2] (in.)	±0.05	Par. 10.3.2.8 Not to exceed ±1 pct	Yes
Control jump[3] (in.)	0.1 (fwd) 0.1 (aft)	Par. 10.3.3.1.1.1 Shall not occur	No
Limit control force[4] (lb)	10	Par. 10.3.3.1.2 20	Yes
Total control travel (in.)	9.3	–	–
Control Dynamics[5]	6 overshoots (fwd) 3 overshoots (aft)	–	–
Trim system lag[6] (sec)	<0.5	–	–
Trim rate (in./sec)	0.4 (fwd) 0.3 (aft)	–	–
Trim authority (in.)	9.2	–	–

NOTES:
[1] Prime Item Development Specification, Sikorsky Aircraft Division, DARCOM-CP-2222-S1000D Part I, 15 October 1979.
[2] Defined as the amount of control travel without corresponding blade travel
[3] Defined as the amount of control movement after trim switch activation following a 20 percent cyclic displacement or a 10 percent pedal displacement
[4] Defined over the total control travel
[5] Defined as the number of oscillations of the control about trim following a 10 percent displacement
[6] Defined as the amount of time following trim activation prior to blade movement

Table 4. Lateral Control System

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Breakout force plus friction (lb)	1.9 (left) 1.5 (right)	Par. 10.3.2.1.1 0.5 min, 2.2 max	Yes
		Par. 10.3.2.1.2 Symmetrical within 10 pct	No
Control Centering	Positive	Par. 10.3.2.2 Positive	Yes
Force versus displacement gradient (lb/in.)	0.9 (left) 0.9 (right)	Par. 10.3.2.2 0.3 min, 0.9 max	Yes
	No discontinuities	Par. 10.3.2.2.1 Positive without discontinuities	Yes
Slope of force gradient for first inch of travel from trim (lb/in.)	1.0 (left) 1.3 (right)	Par. 10.3.2.2.1 Slope for the first inch must be equal to or greater than the slope for remaining control travel.	Yes
Control forces trimmable to zero	Yes	Par. 10.3.2.5 Required	Yes
Control forces maintained at zero	Yes	Par. 10.3.2.5 Required	Yes
Control system freeplay[2] (in.)	±0.05	Par. 10.3.2.8 Not to exceed ±1 pct	Yes
Control jump[3] (in.)	0.7 (left) 0.7 (right)	Par. 10.3.3.1.1.1 Shall not occur	No
Limit control force[4] (lb)	13	Par. 10.3.3.1.2 10	No
Total control travel (in.)	10.2	–	–
Control Dynamics[5]	1 overshoot (left) 1 overshoot (right)	–	–
Trim system lag[6] (sec)	<0.5	–	–
Trim rate (in./sec)	0.4 (left) 0.4 (right)	–	–
Trim authority (in.)	10.0	–	–

NOTES:

[1] Prime Item Development Specification, Sikorsky Aircraft Division DARCOM-CP 2-222-S1000D Part I, 15 October 1979.
[2] Defined as the amount of control travel without corresponding blade travel
[3] Defined as the amount of control movement after trim switch activation following a 20 percent cyclic displacement or a 10 percent pedal displacement
[4] Defined over the total control travel
[5] Defined as the number of oscillations of the control about trim following a 10 percent deplacement
[6] Defined as the amount of time following trim activation prior to blade movement

The force versus displacement gradient was 0.9 lb/in. for both directions of lateral control which was the maximum allowed by the PIDS. All control forces were trimmable to zero and maintained zero control forces at all trim positions tested. The lateral limit control force was 13 pounds which exceeded the requirements of paragraph 10.3.2.5 of the PIDS by 30 percent. Due to priorities set during the A&FC, trim runaways were not evaluated; therefore, the significance of the high limit control forces on a production UH-60A is unknown. Trim runaways should be evaluated during future tests. In flight, the lateral control jump was less (approx 0.2 in.) than that measured on the ground and was not as large as seen in the past (PAE IIA, ref 16, app A). The large lateral control jump was still irritating to the pilot and causes him to make a lateral cyclic control displacement while depressing the trim release switch. As with the longitudinal axis, the lateral control jump failed to meet the requirements of paragraph 10.3.3.1.1.1 of the PIDS. The large lateral control jump (lateral cyclic stick jump) is a shortcoming previously reported (PAE IIA, Dec 1978).

35. Due to the design of the directional control system, very few of the planned control system characteristics test were feasible, but those accomplished are summarized in table 5. The directional control breakout force (plus friction) was 3.5 pounds for both left and right pedal. The directional control forces were trimmable to zero; however, zero control force could not be maintained in ball-centered flight. With the trim system engaged, the right pedal would slowly (approx 0.03 in./sec) drive forward requiring increased foot pressure on the left pedal to maintain trimmed forward flight. The left pedal forces increased approximately 50 pounds during a 40 second interval. The pedal driving appeared to be random in that it occurred above and below 60 KIAS with both pilot's feet on or off the pedal micro switches. The pedal driving would normally drive the aircraft to approximately 1 1/2 ball widths out of trim (coordinated flight); however, the pedals were driven to a control stop in some cases. This phenomenon has been documented in several different UH-60A aircraft. The driving could be stopped by turning the flight path stability (FPS) off or by depressing the trim release switch. As a result of the driving pedals, the pilot found the aircraft was always "out of trim" whenever he was not concentrating on this specific task. The uncommanded directional control input in trimmed flight will significantly increase the pilot's work load (HQRS 7) in instrument meteorological conditions (IMC) and is still a deficiency which was previously reported (PAE III, Apr 1979). The directional control system failed to meet the requirements of paragraph 10.3.2.5 of the PIDS in that zero control force could not be maintained. It should be noted that the test UH-60A was equipped with an "old" AFCS computer (PN 70901-02903-103); therefore, it is possible this problem has been corrected with the "new" AFCS computers (PN 70901-02903-104). The directional control exhibited slight control jump which failed to meet the requirements of paragraph 10.3.3.1.1.1 of the PIDS but the directional control jump was insignificant and not objectionable.

36. The collective control system characteristics are summarized in figure 84 and table 6. This system employs an adjustable friction device that enables the pilot to set the desired breakout force (plus friction). The breakout force, including friction, measured with the adjustable friction OFF was 0.4 pound down and 1.6 pounds up. It was noted throughout the A&FC evaluation that the breakout including friction force with the adjustable friction OFF increased. A later measurement revealed the breakout including friction force with adjustable friction OFF had increased to 4.8 pounds up and 4.2 pounds down. This meets the breakout force plus friction

Table 5. Directional Control System

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Breakout force plus friction (lb)	3.5 (left)	Par. 10.3.2.1.1 3.0 min, 7.0 max	Yes
	3.5 (right)	Par. 10.3.2.1.2 Symmetrical within 10 pct	Yes
Control forces trimmable to zero	Yes	Par. 10.3.2.5 Required	Yes
Control forces maintained at zero	No	Par. 10.3.2.5 Required	No
Control system freeplay[2] (in.)	±0.02	Par. 10.3.2.8 Not to exceed ±1 pct	Yes
Control jump[3] (in.)	<0.1 (left) <0.1 (right)	Par. 10.3.3.1.1.1 Shall not occur	No
Limit control force[4] (lb)	3.5	Par. 10.3.3.1.2 50	–
Total control travel (in.)	4.8	–	–
Control Dynamics[5]	No overshoots	–	–

NOTES:

[1] Prime Item Development Specification, Sikorsky Aircraft Division, DARCOM-CP-2222-S1000D Part I, 15 October 1979.
[2] Defined as the amount of control travel without corresponding blade travel
[3] Defined as the amount of control movement after trim switch activation following a 20 percent cyclic displacement or a 10 percent pedal displacement
[4] Defined over the total control travel
[5] Defined as the number of oscillations of the control about trim following a 10 percent displacement

Table 6. Collective Control System

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Breakout force plus friction (lb)	1.6[2], 4.8[3] (up)	Par. 10.3.2.1.1 1.0 min, 5.0 max	Up, yes
	0.4[2], 4.2[3] (down)		Down, no[4], yes
		Par. 10.3.2.1.2 Symmetrical within 10 pct	No[2,3]
Force versus displacement gradient (lb/in.)	0.1[2] (up) 0.1[2] (down)	None	–
Slope of force gradient for first inch of travel from trim (lb/in.)	0.9[2] (up) 1.7[2] (down)	None	–
Limit control force[5] (lb)	4[2]	Par. 10.3.3.1.2 20	Yes
Total control travel (in.)	9.7	–	–

NOTES:

[1] Prime Item Development Specification, Sikorsky Aircraft Division, DARCOM-CP-2222-S1000D Part I, 15 October 1979.

[2] Adjustable friction at zero at beginning of A&FC.

[3] Adjustable friction at zero near end of A&FC.

[4] Specification should be able to be met by increasing adjustable friction.

[5] Defined over the total control travel

requirement of paragraph 10.3.2.1.1 but fails to meet the symmetrical breakout forces requirement of paragraph 10.3.2.1.2 of the PIDS. Although the breakout forces plus friction were small (0.4 pound down and 1.6 pounds up) when measured during the initial control characteristics evaluation, the collective control system characteristics were satisfactory when the adjustable friction device was set by the pilot.

37. The forward edge of the pilot's or copilot's seat is contoured to accommodate the crotch strap of the pilot restraint system. With the seat full up, full forward and a pilot in the seat, the cyclic control travel is reduced to approximately 4 inches laterally and 6 inches longitudinally. With the seat approximately 50% of full up and full forward the travel is limited to approximately 9 inches laterally and 7.5 inches longitudinally. This situation could easily be experienced when performing a slope landing task (par. 75). The inability to achieve full aft longitudinal cyclic control and the restriction of lateral cyclic control are deficiencies as previously reported (GCT, Nov 1976).

Control Positions in Trimmed Forward Flight

38. Control positions in trimmed (ball-centered) forward flight were obtained in conjunction with level flight performance testing at the conditions in table 2. Figures 85 through 90, appendix E, present the results of these tests.

39. During the control position evaluation it was found that the right pedal would randomly drive forward which increased the pilot work load significantly while attempting to maintain trimmed flight. This deficiency is discussed further in paragraph 35.

Normal Utility Configuration (Level Flight):

40. The variation of longitudinal control position with airspeed during trimmed level flight generally required increasing forward cyclic control with increasing airspeed. Minor nonlinearities were noted; however, the objectionable control reversals noted during PAE III, were not present. At light gross weights, longitudinal control position versus calibrated airspeed was essentially linear at a forward or aft cg position. Only slight differences in longitudinal control positions (less than 0.6 inches) and pitch attitude (less than 3.0 degrees) at airspeeds greater that 100 KCAS were noted with a large cg shift (11.7 inch shift). Figure 85 compares trimmed control positions when gross weight was varied (19,560 and 16,420 pounds). At airspeeds greater than 70 KCAS, the longitudinal position gradient was approximately 0.015 inch/knot at 16,420 pounds and approximately 0.035 inch/knot at 19,560 pounds. Even though the position gradient change was more than double, it was not considered significant since the pilots did not notice the change in flight. Lateral and directional control positions were essentially unaffected by cg or gross weight. The control positions in forward flight (normal utility configuration) are satisfactory.

Normal Utility Configuration Cargo Doors and Gunner Windows Fully Open:

41. Control positions in level flight were evaluated with the cargo doors and gunner windows fully open (fig. 87, app E). Only a forward cg (fuselage station (FS) of 346.8) and a weight of 16,460 pounds was investigated. Pitch attitude trends were

similar to other configurations tested. There were only slight differences in longitudinal, lateral and directional control position trends when compared to a normal utility configuration cargo doors and gunner windows closed. Control positions in trimmed level flight with the cargo doors and gunner windows fully open are satisfactory.

Infrared Reduction Subsystem, XM-130 and, AN/ALQ-144 Configuration:

42. The control positions in trimmed forward flight with the IR reduction subsystem, chaff dispenser set (XM-130), and countermeasure set (AN/ALQ-144) installed were evaluated only at a forward cg (approx FS of 347) and the results are presented in figure 88. Pitch attitude did not appear to be a function of gross weight nor did the pitch attitude change with the installation of the IR reduction subsystem as reported in PAE IIIA (ref 17, app A) (AN/ALQ-144 was not installed during PAE IIIA). At airspeeds less than 70 KCAS and at heavy gross weights (approx 20,000 pounds) the aircraft required approximately 1/2 inch additional right cyclic to maintain trimmed flight when compared to light gross weights (less than 17,000 pounds). Longitudinal control position was not significantly effected by aircraft weight and the trend was similar to that exhibited with a "clean" UH-60A flown at the basic structural design gross weight. The control positions in trimmed forward flight with the IR reduction subsystem, XM-130, and AN/ALQ-144 installed are satisfactory.

Normal Utility Configuration (Climbs and Autorotations):

43. Figure 89, appendix E presents results of the control positions in trimmed MCP climbs and autorotation. The pitch attitude in autorotations was approximately zero degrees throughout the autorotational airspeed range. The pitch attitude in MCP climbs was approximately zero degrees up to 75 KCAS then decreased linearly to 4 degrees nose down at 126 KCAS. The longitudinal control position trends exhibited only small differences in the two modes of flight. The small differences in pitch attitude and longitudinal control position between climb and autorotation significantly reduced pilot work load when performing tasks which require large power changes. In MCP climbs the lateral control positions shifted approximately 0.7 inch to the right. The control positions in trimmed MCP climbs and autorotation are satisfactory.

44. A pitch oscillation occurred during high power climbs near best rate of climb airspeed. At rates of climb in excess of 1700 ft/min and at airspeeds below 70 KIAS the stabilator would fluctuate between 15 and 25 degrees trailing edge down (TEDN). As a result of the stabilator fluctuations, airspeed could only be controlled within ±3 knots even with large longitudinal control excursions (±3/4 inch). At rates of climb below 1700 fpm the stabilator fluctuations were reduced significantly and adequate airspeed control (±1 knot) was possible. The pitch oscillation in low speed flight (50 - 70 KIAS) with high rates of climb (greater than 1700 fpm) is a shortcoming.

Static Longitudinal Stability

45. Collective fixed static longitudinal stability tests were conducted with the pitch bias actuator locked in the centered position at the conditions listed in table 2. Test results are presented in figures 91 and 92, appendix E. The static longitudinal

stability was essentially positive about the trim airspeeds evaluated and is adequate. Comparison of these results with previous test results will be discussed further in paragraph 113.

Maneuvering Stability

46. Maneuvering stability (figs. 93 and 94, app E), was evaluated at the conditions listed in table 2. The method utilized steady-state left and right turns at constant airspeed and collective settings (set at power for level flight). At an average trim airspeed of 118 KCAS, the gradient of control position versus g (stick fixed maneuvering stability) was shallow but positive to 1.4 g. Above 1.4 g, the stick fixed maneuvering stability became negative but was not objectionable within the conditions tested. Some lateral control displacement was required to maintain bank angle; however, it was not objectionable. Maneuvering stability was also evaluated with the pitch bias actuator (PBA) locked and centered (fig. 94) and will be discussed in detail in paragraph 113. The maneuvering stability of the aircraft is satisfactory.

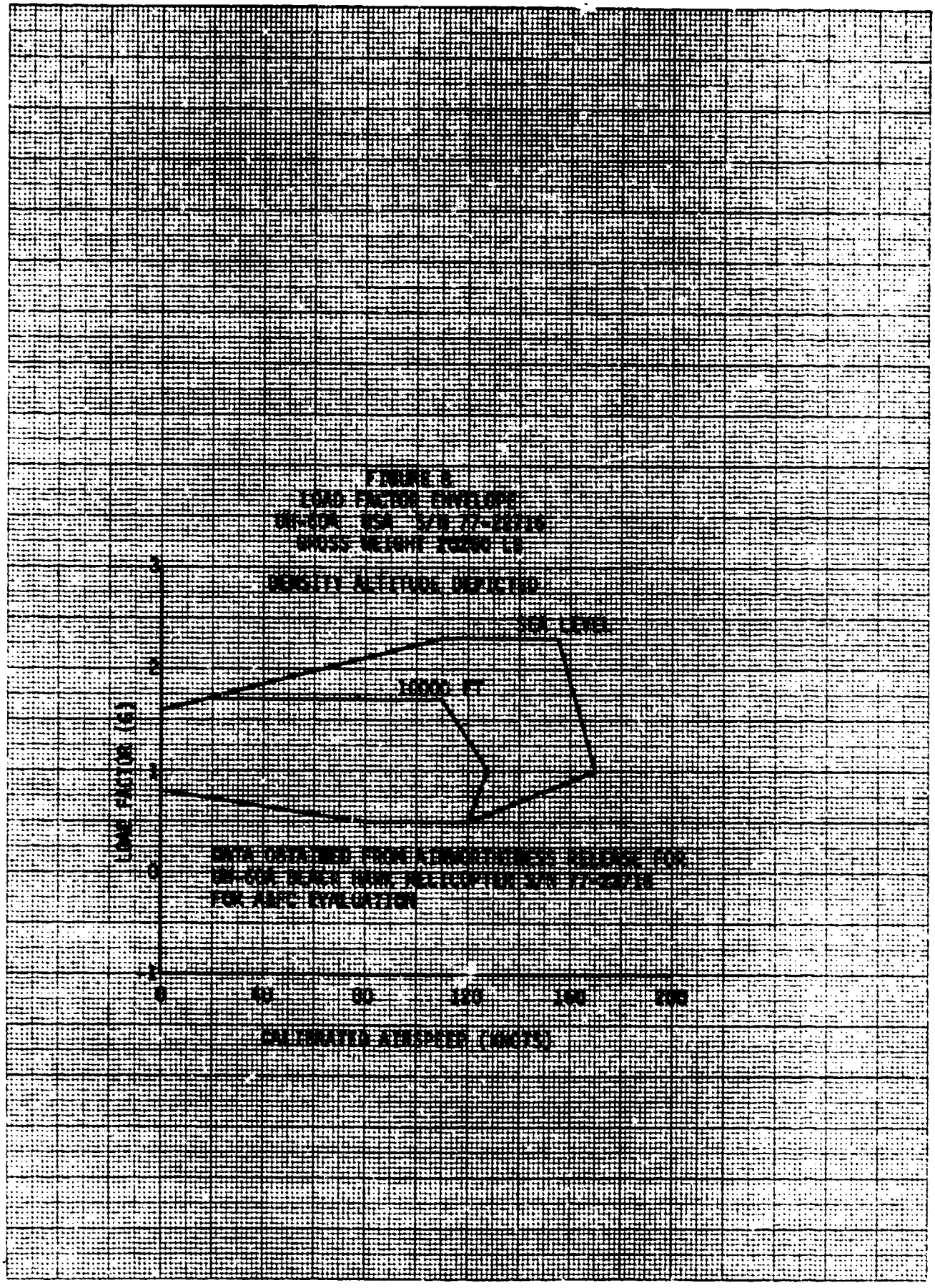
Dynamic Stability (Gust Response)

47. The gust response of the UH-60A was evaluated qualitatively in meteorological conditions ranging from calm to severe turbulence as defined in the Flight Information Handbook (ref 18, app A). The aircraft could be flown "hands off" for extended time periods (greater than 1 minute) in moderate turbulence. Other than transient airspeed and altitude fluctuations, the aircraft AFCS was capable of maintaining long term attitudes with little or no pilot input. The gust response of the UH-60A with AFCS engaged is an enhancing characteristic.

Controllability

48. Controllability tests were conducted in forward flight at 90 and 140 KCAS to evaluate the control power, response, and sensitivity characteristics of the UH-60A. Controllability was measured in terms of aircraft attitude displacement (control power), angular velocities (control response), and angular accelerations (control sensitivity) about an aircraft axis following a control input (step) of a measured size. Following the input all controls were held fixed until a maximum rate was established or until recovery was necessary. The magnitude of the inputs was varied by using an adjustable rigid control fixture. Controllability tests were conducted at the conditions listed in table 2.

49. Longitudinal controllability characteristics are presented in figure 95, appendix E. Neither the magnitude of longitudinal control power (pitch attitude change after 1 second following a 1 inch input) or longitudinal control response (maximum pitch rate per inch control input) varied with airspeed or direction of the inputs. A 4 degree attitude change with a maximum pitch rate of 6 deg/sec was noted. Longitudinal control sensitivity (maximum pitch acceleration per inch control input) appears to vary with airspeed. At 90 KCAS the control sensitivity was 13 deg/sec^2 and at 140 KCAS it was estimated to be 17 deg/sec^2. Roll coupling was evident to the pilot at both airspeeds but did not significantly increase the pilot work load. Due to the limited vertical load factor envelope (fig. B) at V_H, the largest forward input made was 0.6 inch which resulted in a vertical load factor of 0.63 (fig. 96, app E). It should be noted that the minimum allowable vertical load factor of 0.75 at these conditions was inadvertently exceeded as shown in figure 96.


LOAD FACTOR (G)
3
2
1
0
-1
0
40
80
120
160
200
1000 FT
35K LEVEL

In low level or contour flight the pilots will be required to make large forward inputs in order to take full tactical advantage of the terrain. In the present configuration the pilot can unintentionally exceed the load factor limits of the UH-60A with no indication the limit was exceeded. The limited load factor envelope at high gross weights is a shortcoming. Consideration should be given to expanding of the load factor envelope of the UH-60A and incorporating the envelope in the operator's manual. Until the high gross weight load factor envelope is expanded a "g" meter should be installed in the UH-60A.

50. Lateral controllability characteristics are presented in figure 97, appendix E. As with longitudinal controllability, the lateral control power, response, and sensitivity did not change with the direction of input. However, both control response and sensitivity varied slightly with airspeed. The roll attitude change after 1 second was approximately 7 degrees. Slight pitch coupling was noted at both airspeeds but did not increase pilot work load. The control response at 90 KCAS was 10 deg/sec and at 140 KCAS was 11 deg/sec. Control sensitivity varied from 27 deg/sec^2 at 90 KCAS to 30 deg/sec^2 at 140 KCAS. The lateral control response was similar to that found in a previous evaluation. The roll rate would peak at values close to that found in the GCT, but would then decrease to a steady-state rate approximately 4 deg/sec less than the peak (fig. 98). The decrease in roll rate following the initial peak is still a shortcoming which was previously reported (Project No. 77-18, Jan 1978, ref 19, app A).

51. Directional controllability characteristics are presented in figure 99, appendix E. The magnitude of the directional control response did not change with direction of input nor with airspeed. The magnitude of directional control power did change with the direction of input; however, it did not vary with airspeed. The directional control power with a left input was 6 degrees and 7 degrees with a right input. The directional control sensitivity varied slightly with airspeed but did not vary with the direction of input. The directional control sensitivity was 15 deg/sec^2 at 90 KCAS and 24 deg/sec^2 at 140 KCAS. Limited directional controllability was conducted during this evaluation and consideration should be given to performing additional tests.

52. During the A&FC evaluation, the cyclic was inadvertently bumped which resulted in an abnormal aircraft pitch response. This abnormal pitch response was evaluated by releasing the longitudinal cyclic control after it was displaced from a trimmed position. This resulted in the cyclic control abruptly returning to the trimmed position which resulted in a jarring aircraft pitch response that was sensed by the pilots as a high vertical acceleration. The large, sudden accelerations that resulted from the abrupt cyclic return is very disconcerting to the crew. This jarring pitch response is of such magnitude that it may cause the pilots to assume they have structurally damaged the aircraft. Based on AVRADCOM's comments to PAE III (ref 20, app A), the jarring pitch response will not cause structural damage. The jarring pitch response to a cyclic release following a longitudinal cyclic displacement against trim has been downgraded from a deficiency, as previously reported (PAE III, Apr 1979), to a shortcoming.

Ground Handling Characteristics

53. The ground handling characteristics of the UH-60A, which included starting, systems checks, taxiing, and engine/rotor shutdown, were evaluated concurrently

with other tests. The starting and shutdown procedure for the UH-60A were fairly straightforward; however, it was felt the starting checklist was excessively long. For example, the "Flight Control Hydraulic Check" was 10 or 16 steps long (depending on which type of hydraulic servos were installed).

54. Both engines on the UH-60A can be started simultaneously. This capability will significantly reduce turnaround time. Additionally, missions requiring low "scramble" times such as medivac or combat assaults will be greatly enhanced. The capability of simultaneous dual engine starts is an enhancing characteristic.

55. During the engine starting sequence the Hydraulic Leak Test System was checked as required in paragraph 8-26-14 (ref 4, app A). When the HYD LEAK TEST switch was placed in the TEST position, several caution lights illuminate. Additionally, by moving the cyclic, the TRIM ACT latch on the computer maintenance indicators will latch, indicating a trim failure. Due to the logic of the computer maintenance indicators the TRIM ACT will not reset when the HYD LEAK TEST switch is reset. The indicators will now show that the trim has failed when the crew chief checks the computer maintenance indicators. Therefore, the trim maintenance indicator will be of little value as a maintenance guide. The erroneous TRIM ACT indication on the maintenance panel following the hydraulic leak test system check is a shortcoming.

56. Large overshoots of N_P and main rotor speed (N_R) occurred during the engine overspeed test required in the run-up procedures. The overshoots occurred during either engine system check when the two overspeed test buttons were released. The overshoot resulted in rotor speeds from 101 to 105 percent. The test procedure was varied slightly by releasing the test buttons at various transient rotor speeds; however, the rotor surge could not be reduced with any repeatability. The large N_P and N_R overshoots that occur with the release of the engine overspeed test buttons is a shortcoming previously reported (PAE III, Apr 1979).

57. The UH-60A was designed with a tail wheel which could be locked or unlocked from the cockpit. During the evaluation a new design tail wheel locking device was installed. The new design was an attempt to eliminate problems previously encountered in locking and unlocking the tail wheel. Though the design did significantly reduce the frequency of the above mentioned problem, an intermittent locking and unlocking problem still exists. The pilot had to manipulate the pedals to relieve pressure on the tail wheel locking pin to effect engagement or disengagement. When difficulties were experienced in unlocking the tail wheel, an excessive ground roll was usually required prior to achieving an unlocked condition. The difficulty in unlocking and locking the tail wheel, even with the new design, remains a shortcoming previously reported (GCT, Nov 1976).

58. The position of the locking pin can normally be determined by lights collocated with the tail wheel switch. However, due to the logic of the tail wheel locking system, all lights will extinguish when commanded to transition between lock to unlock or vice versa. Therefore, if the problem discussed in the previous paragraph occurs and the tail wheel switch is recycled, the pilot will not know if the tail wheel switch has been commanded to go to a locked or unlocked mode. If the difficulty in unlocking and locking the tail wheel (para 57) were corrected, this would cease to be a problem. The lack of an indication of the commanded tail wheel locking pin position is a shortcoming.

59. During ground taxi on a level hard surface with collective settings sufficient for the aircraft to be "light" on its wheels, a self exciting aircraft pitch oscillation was experienced (fig. C). The oscillation was neutrally damped and would stop when the collective was lowered. The amplitude of the oscillation was significantly reduced with both stability augmentation systems off or with the stabilator in trailing edge up position. The pitch oscillation was more frequently observed at lighter gross weights. The aircraft pitch oscillation is a shortcoming previously reported (PAE IIA, Dec 1978).

60. During ground handling evaluations the UH-60A exhibited a divergent mechanical instability (fig. D). The instability occurred when the collective control was increased to the point where the aircraft was "light" on its wheels (collective greater than 30 percent), the longitudinal cyclic placed in an aft position (greater then 75 percent from full forward) and the wheel brakes set to prevent aircraft movement. The mechanical instability appeared to be dependent upon aircraft cg and gross weight; therefore, position of the cyclic and collective controls where initiation occurs may vary. The instability was a self excited oscillation (approx 3 Hz) and the rate at which the amplitude increased was slow enough to allow the pilot to damp the oscillation by lowering the collective or by a forward cyclic input. The mechanical instability appears only in the roll axis. The self excited divergent mechanical instability during ground operation is a shortcoming previously reported (PAE IIIA, Dec 1979). Consideration should be given to incorporating the following caution in the UH-60A operator's manual:

CAUTION

The UH-60A has encountered ground resonance when the following three conditions occur simultaneously:

1) Parking brake ON

2) Aircraft "light" on the wheels

3) Extreme aft cyclic control

The ground resonance is characterized by a lateral oscillation (approximately 3 cycles per second). During these oscillations only small attitude changes are encountered; however, roll rate will increase significantly within two seconds. The resonance can be quickly damped by reducing collective, or by bringing the aircraft to a hover, or by centering the longitudinal cyclic control.

Takeoff Characteristics

61. Takeoff characteristics were evaluated in conjunction with other tests throughout the test program. Additional takeoff tests were performed to evaluate aircraft pitch-over during takeoff and acceleration at the conditions in table 2. Characteristic takeoff time histories are presented in figures 100 and 101, appendix E. Takeoff characteristics at heavy gross weight (21,760 pounds) were similar to those of lighter gross weights. A normal takeoff, acceleration, and

FIGURE 6.
[illegible] DURING TAXI
[illegible]

GROSS WEIGHT (LB)	LONG C.G. (IN)	LAT C.G. (IN)	DENSITY ALTITUDE (FT)	OAT (°C)	MAIN ROTOR SPEED
15,830	[illegible] MID	0.0 MID	2040	24.5	258

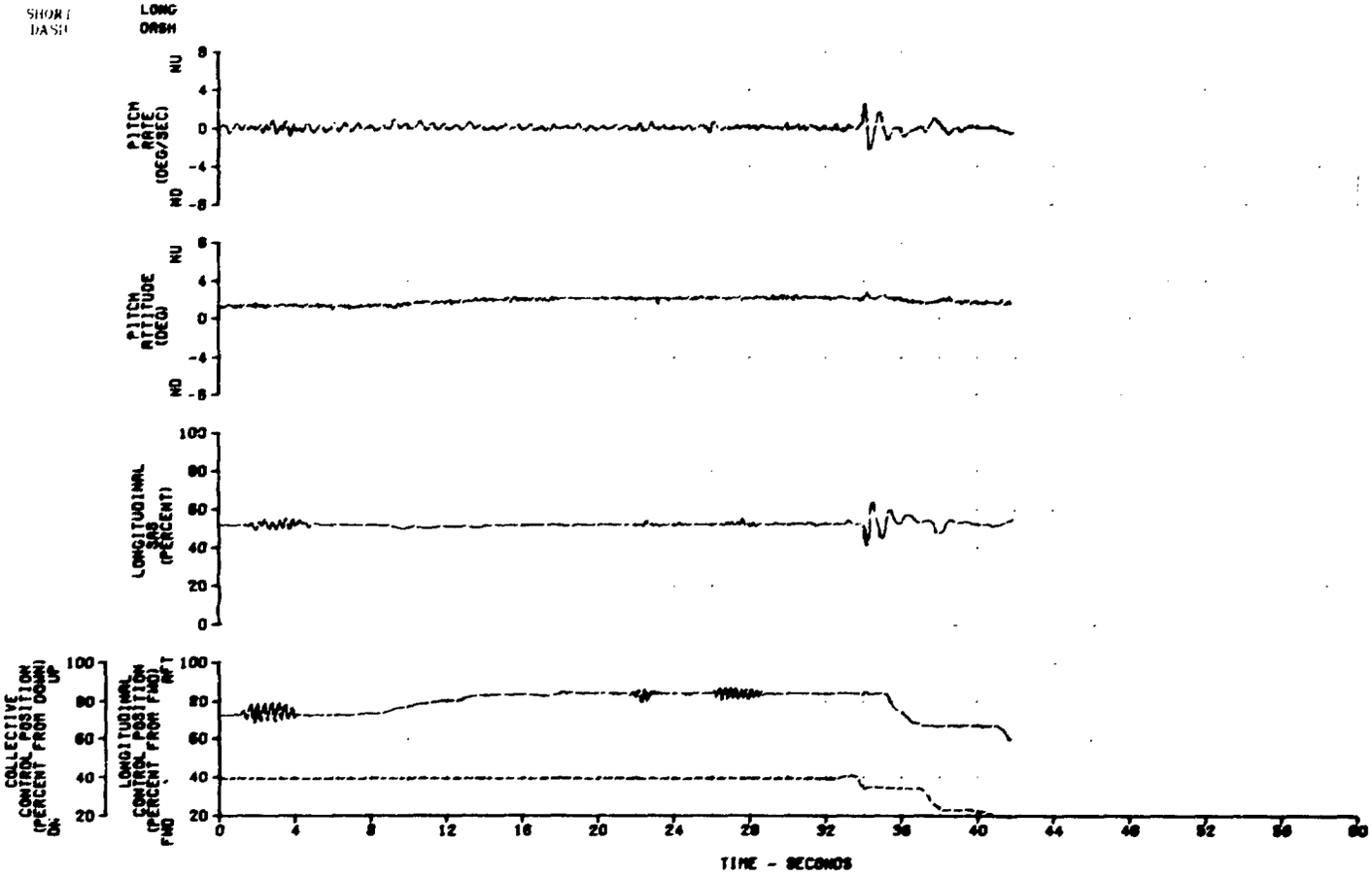
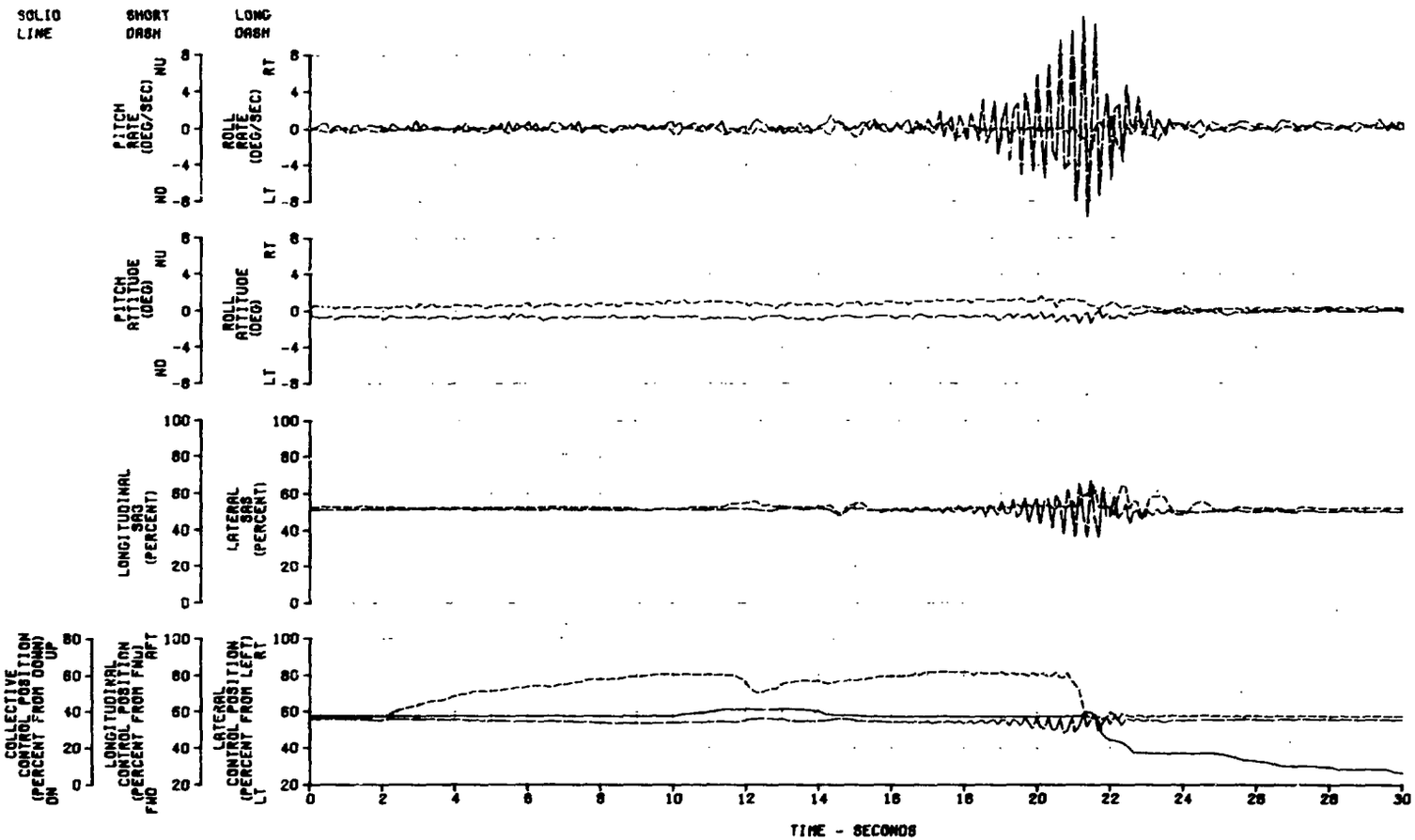


FIGURE D
MECHANICAL INSTABILITY
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (° C)	MAIN ROTOR SPEED
15,900	351.0(MID)	0.0(MID)	2060	25.5	258



climbout was performed from a 2-foot hover height. The technique used was to increase torque approximately 6 percent during a level acceleration. At 25 KTAS, the aircraft was rotated to a normal climb pitch attitude. The technique used from this point throughout the remainder of the takeoff was either (1) maintain fixed longitudinal cyclic throughout the climbout to demonstrate the aircraft pitchover or (2) maintain a constant pitch attitude throughout the climbout to demonstrate the longitudinal cyclic movement required to perform this task.

62. A representative time history of a normal takeoff from a hover with longitudinal cyclic fixed after establishing a climb attitude at approximately 25 KTAS is presented in figure 100. Stabilator programing with increasing airspeed was essentially linear from 35 to 70 KIAS and accompanied by a constant pitch attitude as indicated from 12 to 20 seconds into the takeoff. However, during the following 1.3 seconds the indicated airspeed dropped from 70 to 55 KIAS causing the stabilator to change its direction of travel and program from 18 degrees to 22 degrees TEDN. This stabilator angle change due to indicated airspeed fluctuations was accompanied by a nose down pitching moment which resulted in an additional 5 degrees nose down pitch attitude. The excessive nose low attitude caused the aircraft to descend from 74 feet to 14 feet above ground level in 7 seconds. In this instance no aft longitudinal cyclic input was required to prevent the aircraft from making ground contact. However, some takeoffs using this technique did require the pilot to recover.

63. A representative time history of a normal takeoff from a hover attempting to maintain a constant climb attitude during the climbout phase is presented in figure 101. Stabilator programing with increasing airspeed was essentially linear from 35 to 60 KIAS and was accompanied with a constant pitch attitude requiring no longitudinal cyclic inputs from 12 to 20 seconds into the takeoff. During the next 3 seconds, the airspeed dropped from 60 to 40 KIAS accompanied by a change in stabilator direction of travel from 23 degrees TEDN to 26 degrees TEDN. The resulting nose down pitching moment required a 1.2 inch aft cyclic movement to arrest pitch rate and maintain a constant climb angle which is contrary to normal longitudinal control displacement when performing this maneuver (HQRS 7). The indicated airspeed then began to increase linearly with time from approximately 40 to 80 KIAS with the stabilator programing linearly. This pitchover characteristic during takeoff in IMC, night, and during sling load operations will greatly increase the pilot's work load and could result in aircraft damage and/or personnel injury. The severity of the pitch over is random and not repeatable and many worse instances than that depicted in figure 101 were experienced. The nose-down pitching moment experienced during takeoff and climbout is a deficiency previously reported (PAE III, Apr 1979).

64. Takeoffs were also performed with the stabilator in the manual mode at 0, 5, 10, and 15 degrees TEDN. The longitudinal cyclic position was fixed at a climb pitch attitude when accelerating through approximately 25 KTAS. A nose down pitching moment resulting in a nose down pitch attitude was observed during the takeoff acceleration for manual stabilator angles of 5, 10, and 15 degrees TEDN with less pitchover occurring as the manually selected stabilator angle approached 0 degrees TEDN. The pitchover which occurred with the longitudinal cyclic held fixed and the stabilator in the automatic mode (fig. 100) did not occur under similar conditions with the stabilator in the manual mode at 0 degrees TEDN. The forward cyclic displacement from hover required to obtain the desired climb pitch attitude at

approximately 25 KTAS was greater (1.5 inches) at the 0 degrees TEDN stabilator angle than the takeoffs performed with the stabilator in the automatic mode (0.7 inches) although not objectionable.

65. Pilot work load to establish the climb pitch attitude with the stabilator in the manual mode at 0 degrees TEDN (HQRS 4) was slightly greater than that required with the stabilator in the automatic mode (HQRS 3) due to the larger forward cyclic displacement required to obtain the climb pitch attitude at 25 KTAS. Pilot work load required to maintain a constant climb angle throughout the acceleration, past 25 KTAS, was much less with the stabilator set at 0 degrees TEDN in the manual mode (HQRS 2) than was the pilot work load with the stabilator in the automatic mode (HQRS 7). This was due to the reduction of the longitudinal cyclic control reversals required to maintain the constant climb angle and pitch attitude. The airspeed indicator fluctuations were still apparent while performing a takeoff with the stabilator in the manual mode and set at 0 degrees TEDN, but the pitchover characteristic was not observed.

66. Airspeed indications were erratic during accelerations from hover to 60 KIAS. Airspeed changes as large as 20 knots in less than 1 second were observed. Rapid transitions through the low airspeed range decreased the severity of the fluctuations in airspeed indications. The erratic airspeed indications during aircraft acceleration below 60 KIAS reduces the useability of the airspeed indicators in the low airspeed range and is a shortcoming. The erratic airspeed indications observed during PAE III, April 1979, was reported as a deficiency; however, both the oscillations and magnitude have been reduced significantly. Therefore, the previously reported deficiency has been downgraded to a shortcoming.

67. On several occasions the heading hold feature of the FPS failed resulting in large yaw displacements during takeoffs. There was no cockpit indication of the failure prior to the yaw excursions. The failure characteristics varied from a slow nose left to a moderate nose right yaw. The random nature of the heading hold failures would delay the pilot's reaction to the yaw excursions and could result in a dangerous flight condition. Since the heading hold feature is so unreliable, the pilot will lack sufficient confidence in the system to fly the aircraft with his feet off of the pedals, thereby reducing system utilization. The unreliability of the AFCS heading hold and subsequent large yaw excursion is a deficiency previously reported in PAE III, April 1979). AVRADCOM (ref 20, para j) stated the problem was in the heading hold logic of the AFCS computer and a modification was instituted. USAAEFA pilots who have flown several of Fort Rucker's UH-60A's recently (May 1981) noted the problem is still prevalent.

68. To use the heading hold feature below 60 KIAS the pilot had to position his feet so that the pedals were free to move without interference. This was annoying and significantly reduced the usefulness of the heading hold feature. If the pilot were to inadvertently hit the pedal while making power changes (*i.e.*, takeoffs with sling loads) the heading hold would disengage and result in large yaw excursions. Considering the unreliability of the heading hold subsystem (para 67), it will be common practice to "guard" the pedals. This will result in inadvertent disengagement of heading hold since heading hold drives the pedals. The requirement for the pilot to consciously reposition his feet to engage heading hold (below 60 KIAS) is a shortcoming previously reported (Project No. 77-18, Jan 1978). A redesign of the heading hold subsystem should be considered.

69. Large lateral control displacements were required to maintain a straight ground track during takeoffs and landings. On transition from hover to forward flight, left lateral cyclic displacements (approx 1-1/2 inches) were required to maintain the desired takeoff ground track. Figure 103, appendix E shows this trend clearly, even though the points were taken during static conditions. In transition from forward flight to a hover, right lateral cyclic was required to maintain a straight ground track to the point of intended touchdown. The large lateral control displacement increased the pilot work load in maintaining the desired ground track during takeoffs and landings (HQRS 4) and is a shortcoming previously reported (PAE III, Apr 1979).

70. In addition to the random trim failures discussed in paragraph 146, it was found that the trim failed consistently during high performance level accelerative takeoffs. The failure occurred when the longitudinal cyclic was sharply pulled aft to initiate a "cyclic climb" preceded by a pushover. Level accelerative takeoffs could not be accomplished without the trim failing. The failure of the trim system during level accelerative takeoffs is a shortcoming. The above shortcoming appears similar to the shortcoming previously reported in Project No. 77-18, January 1978, and PAE IIA, December 1978.

Slope Landing Evaluation

71. The slope landing capabilities were evaluated on aircraft S/N 77-22717 at the test conditions listed in table 2. The landings and takeoffs were made on measured, soil stabilized slopes ranging from 10 to 15 degrees. The actual slopes between the gear alighting points was measured after the aircraft departed the slope. The main and tail wheel struts were serviced in accordance with technical manual (ref 21, app A) prior to conducting the test.

72. The technique employed during landing and takeoff was essentially the same for each slope orientation tested. The parking brake was set and the tail wheel locked during the tests. Coordinated cyclic, collective, and directional pedal inputs were required until the helicopter was firmly positioned on the slope. For left and right wheel up-slope landings, the main gear and tail wheel contacted the ground almost simultaneously. Nose up and down slope landings required less pilot effort since roll attitude control was not as demanding as it was in the left and right wheel up landings. The aircraft attitudes were measured on the cabin floor using an inclinometer after the aircraft was firmly positioned on the slope. The difference between aircraft attitude and slope angle was due to differential compression of the gear struts. The aircraft attitude for each slope orientation is presented as follows:

Slope (deg)	Aircraft Attitude (deg)
14.0 right wheel up	17.0 right wheel up
13.0 left wheel up	15.0 left wheel up
11.5 nose up	13.5 nose up
11.5 nose down	14.0 nose down

73. The aircraft was controllable throughout the landing and takeoff on the 12 degree left and right wheel up slope with adequate control margins remaining in all areas. After the downhill main wheel contacts the ground, slight roll oscillation may be unintentionally induced by the pilot but can be quickly damped by lowering the collective approximately 1 inch (HQRS 4). A left gear up landing and takeoff were made on a 13 degree slope; however, the aircraft was not under full control during landing until the downhill wheel made ground contact. With full left cyclic, the aircraft slid downhill until the collective was lowered sufficiently to cause the right wheel to make ground contact. After full lateral cyclic was applied, the slide distance was a function of how rapidly the pilot reduced collective until the downhill wheel made ground contact. A right wheel up landing and takeoff were performed on a 14 degree slope. Full right lateral cyclic was used, but the aircraft did not slide down the hill on landing or takeoff as with the left wheel up landing.

74. The aircraft was controllable throughout the 10 degree nose up landing and takeoff (HQRS 3). When attempting a nose up landing on a slope in excess of 12 degrees, the parking brake would not prevent the main wheels from turning thus allowing the aircraft to roll down the hill. The maximum slope measured for nose up landing was 11.5 degrees with parking brakes effectively preventing the aircraft from rolling down the hill. Full forward cyclic was used during this landing and takeoff.

75. The UH-60A was controllable throughout the 10 degree nose down landing and takeoff with adequate control margins remaining in all axes (HQRS 3). A nose down landing on a 11.5 degree slope was performed. However, with full aft cyclic, the aircraft rolled downhill on the tail wheel until the main wheels made ground contact. After full aft cyclic was applied, the distance the aircraft rolled down the 11.5 degree slope was a function of how rapidly the collective was lowered. When main wheels made ground contact, the parking brakes held, and the aircraft came to a stop.

76. A separate IGE hover flight over level ground was performed to determine control position requirements with the test aircraft ballasted at asymmetric cg loading and near the forward and aft cg limit. The test gross weight was 16,400 pounds, 4240 feet density altitude and 258 rotor rpm. The results are presented in table 7. If the slope landing and takeoff testing was defined as the base line cg loading (mid longitudinal, mid lateral), the aircraft required 0.8 inches more right lateral stick at a butt line (BL) -3.9 and 0.7 inches more left lateral cyclic control at a BL 3.8. Likewise, 0.5 inches more aft longitudinal cyclic control was required with the cg at FS 346.2 and 0.6 inches more forward longitudinal stick with the cg at FS 359.6. During a takeoff or landing operation from a sloped surface at a critical condition (*i.e.*, right wheel up, left lateral cg), less control will be available to prevent the aircraft from sliding. Additional slope landing evaluations should be performed at the extreme cg locations to define the control margins under these conditions.

Table 7. IGE Hover Control Positions At Various CG Loadings

CG Loading (inches)	Longitudinal (inches from full fwd)	Lateral (inches from full left)	Directional (inches from full left)	Collective (inches from full down)
FS 352.7 (mid) BL 0.0 (mid)	6.4	5.3	1.7	5.9
FS 352.6 (mid) BL -3.9 (lt)	6.5	6.1	1.7	5.8
FS 352.9 (mid) BL 3.8 (rt)	6.3	4.6	1.7	5.8
FS 346.2 (fwd) BL 0.0 (mid)	6.9	5.5	1.6	5.9
FS 359.6 (aft) BL 0.0 (mid)	5.8	5.1	1.7	6.0

77. During slope operations, vertical clearance between the main rotor tip path plane and the ground was extremely low on the uphill side of the helicopter. During testing on the 10 to 15 degree slopes, the tip path to ground clearance was approximately 4 feet on the uphill side. This tip path clearance is a hazard to personnel in close proximity to the helicopter during slope operations. The following WARNING should be included in chapter 8 of the operator's manual:

WARNING

> During slope operations, vertical clearance between the main rotor tip path plane and the ground is extremely low on the uphill side of the helicopter. Personnel must be warned not to approach or depart the helicopter in the uphill direction.

78. The UH-60A failed to meet the slope landing capabilities of paragraph 10.3.7.7.2 of the PIDS (ref 3, app A) in that aircraft control could not be maintained for left wheel up and nose up landings to the 15 degree slope nor for the 12 degree nose down slope.

79. Paragraph 5-31-2 of the operator's manual should be changed to reflect the maximum slope landing that should be attempted with winds of 3 knots or less is 12 degrees left or right wheel up and 10 degrees nose up or nose down.

Low Speed Flight Characteristics

80. The low speed flight characteristics of the UH-60A were evaluated at Coyote Flats and Edwards Air Force Base at the conditions listed in table 2. Aircraft serial numbers 77-22717 and 77-227-16 were used in this evaluation. Aircraft 77-22717 was configured with the IR suppressors, XM-130 chaff dispenser, AN/ALQ-144 IR jammer, and cargo hook. The low speed flight performed at Edwards Air Force Base

was in test aircraft 77-22717. Low speed flight was evaluated on aircraft 77-22716 in the normal utility configuration (standard engine cowling without the XM-130, AN/ALQ-144, and cargo hook) at Edwards Air Force Base and Coyote Flates test sites. Surface winds were 3 knots or less and a ground vehicle and a radar speed gun was used as a pace reference. The low speed flight test data are presented in figures 102 through 110, appendix E.

81. During steady low speed flight, adequate control margins remained in all axes. Directional control margin reached its minimum value of 0.5 inches left pedal remaining (10 percent) at a relative wind azimuth of 60 degrees (critical azimuth) and 35 KTAS under the test conditions at Coyote Flats. The directional control margin at the critical azimuth remained essentially constant at 10 percent from 20 KTAS to the sideward flight limit (35 KTAS). The variation of control positions in low speed fligight showed varying gradients in all axes, depending on the direction and magnitude of the relative wind azimuth; however, these variable gradients did not present a control problem. The variation of control positions observed on aircraft 77-22717 in the IR configuration were similar to those on aircraft 77-22716 in the normal utility configuration. The vibrations for the fourth harmonic of the main rotor (4/rev) in both configurations increased between 10 and 15 KTAS (VRS 4) at all relative wind azimuths tested. The largest longitudinal and lateral control position reversals occurred in the normal utility configuration at a relative wind azimuth of 345 degrees and 30 KTAS but were insignificant. A lateral shuffle (acceleration in the lateral axis) was observed between 285 and 345 degrees relative wind azimuth of 345 degrees and 30 KTAS but were insignificant. A lateral shuffle (acceleration in the lateral axis) was observed between 285 and 345 degrees relative wind azimuth at speeds between 15 and 25 KTAS. The most severe shuffle occurred at the 315 degree azimuth between 17 and 25 KTAS. The most severe shuffle occurred at the 315 degree azimuth between 17 and 22 KTAS in both aircraft configurations; however, the shuffle observed in aircraft 77-22717 in the IR configuration was of less amplitude than 77-22716 in the normal utility configuration. The lateral acceleration (shuffle) was highly damped with an inconsistent frequency from 1 to 2 Hz and the pilot was unable to counter the shuffle with control inputs. This lateral shuffle could cause unnecessary side load on the landing gear when performing a landing from a hover and a wind velocity of 15 to 25 knots from an azimuth of 285 to 345 degrees. The lateral shuffle in low speed flight at relative wind azimuths between 285 and 345 degrees at speeds of 15 to 25 knots is a shortcoming. The following CAUTION should be included in chapter 8 of the operator's manual:

CAUTION

> Landing from a hover with a relative wind azimuth between 285 and 345 degrees with wind velocity or gusts between 15 and 25 knots should be avoided due to the possibility of encountering a lateral shuffle.

Power Management

82. Engine torque matching characteristics were qualitatively assessed throughout the A&FC. The maximum torque difference noted throughout static and dynamic testing was 3 percent, which only occurred at low power settings. At stabilized

power settings greater than 70 percent, the maximum torque difference was one percent. The effective torque matching capability of the T700-GE-700 engines is an enhancing feature.

83. USAAEFA completed an engine assessment December 1977 on Project No. 77-58 (ref 22, app A). There have been several significant modifications to the engines and the various engine controls since that time. The hydromechanical unit (HMU) replacement discussed in paragraph 85 is one such change. A similar engine assessment, as conducted on Project No. 77-58, should be performed on a current production UH-60A.

84. The UH-60A has a TGT limiter which functions as an engine overtemperature protection device. The TGT limiter is set at an activation threshold of 840° ±5° C. The T700-GE-700 engine has a 30 minute TGT limit of 850° C and a 12 second transient limit of 886° C. With the TGT limiter set at 840° C ±5° C the pilot is restricted from fully utilizing the engine power available. With a single engine failure in a performance critical situation, the ability to use the range of engine power up to the 12 second transient temperature power limit may allow the pilot to make a controlled landing. The TGT limiter should be automatically disabled for single engine operation.

85. The emergency power electronic control unit ((ECU) lockout) management characteristics were qualitatively evaluated in level flight at 100 KIAS with the HMU, part Nos. 4046T52G01 and 4046T52G04. Emergency (manual) control of the T700-GE-700 engines is obtained by moving the desired engine power control lever (PCL) to ECU lockout position (full forward). After gaining manual control of the engine, the PCL is retarded quickly to manually adjust engine power to the desired level. To bring the engine out of ECU lockout, the PCL is retarded to the idle detent. When attempting to control engine power (ECU lockout) with the -01 HMU installed, it was not possible to reduce the power of the manually controlled engine below the automatic governed engine, which resulted in high rotor speed. When the pilot attempted to reduce the power, the PCL would go into the idle detent which would signal the engine to go back into the automatic mode. To make the problem worse, when in the manual mode, the position of the PCL was dependent on collective position (the higher the collective the further aft the PCL). During high powered flight, the pilot may not be able to position the PCL far enough aft to avoid a steady state rotor overspeed. Other than in a training flight, while practicing emergency procedures, the pilot will not be operating in ECU lockout; therefore, will not be aware of the problems in this mode. The inability to control rotor speed in the ECU lockout mode with the 4046T52G01 HMU is a deficiency. It should be noted that it appears the 4046T52G04 HMU has corrected this problem.

86. When transitioning from low power flight (torque less than 25 percent) to normal cruise, the main rotor drooped approximately 4 percent. The main rotor rpm returned to the trim value (100 percent) in approximately 3 seconds. The rate of collective application did not appear to significantly affect the main rotor droop. At torque settings greater than 25 percent there was no significant rotor droop. During a maneuver which required low power, such as a steep approach, the large rotor droop was a safety concern when "adding power at the bottom" for deceleration. The poor engine droop characteristics degrade the aircraft's maneuverability and is a shortcoming previously reported (PAE III, Apr 1979).

87. During the A&FC evaluation, engines were inadvertently shut down. Four incidences of engine shutdown occurred during the program with three of them occurring while performing a health indicator test check and one time during a recovery from ECU lockout operation. In all cases, the cam on the PCL did not fall into the idle detent position and the power lever was pulled aft of the detent position resulting in an engine shutdown. The design of the idle detent cam on the engine control quadrant which allows inadvertent engine shutdowns is a deficiency.

88. During this test it was noted that the rotor rpm indicated on the ship instruments always indicated 1 percent lower than the calibrated rotor tachometer. The same problem occurred during PAE III and IIIA with a different aircraft (S/N 77-22714) and a different instrumentation system. At the time, it was considered to be an anomally in aircraft S/N 77-22714. The 1 percent error in the ship rotor tachometer was not of any significance to the pilots; however, the long term effect of flying the aircraft at lower than design rotor speed is an unknown. The UH-60A rotor tachometers should be corrected to read the proper rotational speeds.

Instrument Flight Characteristics

89. Instrument flight characteristics were qualitatively evaluated during various phases of the A&FC program. Takeoff, enroute tasks, and approaches were conducted utilizing all modes of the CIS with the exception of the "back course" mode. A dedicated IFR flight evaluation was not conducted due to time constraints, but qualitatively it appears that the UH-60A does not exhibit any significant handling quality problems when operating in IFR conditions. However, there are several navigational problems associated with the CIS, as discussed below, which if corrected would enhance the capability of the aircraft to fly in IFR conditions.

90. The altitude hold mode was evaluated by conducting climbs and descents at 500 and 1000 ft/min at a trim airspeed of 80 KIAS. Once established in a steady state condition "ALT" was engaged and the pilot followed all control cues displayed on the vertical situation indicator (VSI). In climbs, the maximum altitude overshoot was 140 feet and after one overshoot the altitude would be stabilized within 60 feet of the target altitude. When engaging the "ALT" hold mode during descents, overshoots of 240 feet were observed; however, after one overshoot the altitude would be stabilized within 20 feet of the target altitude. The cues and accuracy provided by the "ALT" mode of the CIS are satisfactory.

91. The heading mode of the CIS was evaluated by setting off-course headings (without using radio navigation aids) and utilizing the "heading set knob." The aircraft was then flown based on commands initiated by the VSI. By following the commands, the aircraft could be flown accurately to the commanded heading. The operation of the heading mode (without radio navigation aids) was satisfactory.

92. The navigation (NAV) mode has four submodes: VOR NAV, ILS NAV, DPLR NAV and FM NAV. All modes except FM NAV were qualitatively evaluated. Due to the electromagnetic interference (EMI) problems between the No. 1 FM radio and automatic stabilator programming, the No. 1 FM radio had to be removed from the UH-60A aircraft and made the FM NAV mode inoperative. The inability to use the No. 1 FM radio due to EMI problems is a shortcoming.

93. During the evaluation of the DPLR NAV submode it was noted that the present doppler configuration only displays distances and ground speeds in kilometers and kilometers/hour respectively. As a result, the pilots must manually convert distance and ground speed into nautical miles and knots to make the doppler information compatible with sectional aeronautical charts and the airspeed indicators. The inability of the DPLR NAV submode of the CIS to display distances and ground speed in units of nautical miles is a shortcoming.

94. The VOR NAV mode incorporates a station passage submode (more commonly referred to as heading lockout). The only function of the heading lockout submode is to eliminate the oscillations in the roll command bar when operating in the "zone of confusion" around a very high frequency omni-directional range (VOR). As the aircraft enters the "zone of confusion" the rapid changes in the lateral deviation indicator (1 degree per second for more than 1 second) causes the CIS to provide roll steering commands to track the course selected on the omni-bearing selector (OBS). The CIS Processor adds the last crab angle correction used to the course chosen on the OBS to give roll steering commands. This mode remains in effect for approximately 30 seconds after the last deviation is sensed. The above mentioned oscillations have been eliminated; however, several significant problems were caused by the heading lockout mechanism.

95. When holding at a VOR station, the UH-60A pilots cannot fully utilize the primary aircraft flight instrument, VSI, for navigation within the holding pattern. At low altitudes of approximately 2000 ft above ground level (AGL), the 'zone of confusion' has a small diameter, therefore the heading lockout would be engaged less than 15 seconds prior to crossing the VOR. When the aircraft leaves the 'zone of confusion' somewhere during the outbound turn, the VSI roll bar will display valid information. At altitudes greater than approximately 4000 ft AGL (altitudes will vary depending on the diameter of the 'zone of confusion'), the aircraft will initially enter the 'zone of confusion' much earlier. Therefore, when the aircraft is in its inbound turn (once established in a holding pattern) the 'zone of confusion' will be re-entered. The roll bar will then command the pilot to fly the heading the aircraft was on when it entered the 'zone of confusion'. With these erroneous roll commands, the VSI may indicate to the pilot that he is on course when in fact, he is not and is flying further off the inbound course. At higher altitudes (estimated to be greater than 8000 ft AGL), the entire holding pattern will be flown in the 'zone of confusion'. This will result in the roll bar commanding the pilot to fly the original inbound heading and not the inbound course of the holding pattern. Since there are no heading lockout engagement cues, the pilot will not know when the VOR NAV mode is in the heading lockout submode. This compels the pilot to ignore his primary flight instrument. The inability of the pilot to utilize the VSI (roll bar) of the CIS during holding patterns at a VOR is a shortcoming.

96. The heading lockout submode of the CIS is automatically engaged by sensing the rate of change of the lateral deviation indicator. It cannot sense whether the deviations are caused by the "zone of confusion" or by pilot inputs in the course set knob. Therefore, the lockout occurs when the pilot turns the course set knob to select a new course, *i.e.*, change of radial to track inbound or outbound on a different radial, intersection holding, or identification of intersections on final approach which may serve as a final approach fix. This problem is further complicated by information displayed by the roll bar. The heading which the roll bar commands the pilot to fly after the course set knob was moved was not predictable. The movement of the course set knob causes the lateral deviation

indicator to displace. The instant that the lateral deviation indicator moves at a rate of 1.0 deg/sec for more than one second, the roll bar will command a heading to the course displayed on the "course set display" in addition to the original wind correction. For example, suppose the original course was 90° with a required 100° reading to maintain course. Assume a new course of 210° was required and the pilot uses the course set knob. As he turns the course set knob, assuming he exceeds the 1 deg/sec as the "course set display" goes through 110°, he will then be commanded by the roll bar to fly a new heading of 120° (90° away from the desired course) for a minimum of 30 seconds. Thus the pilot must revert to the secondary flight instruments, which are of no improvement over the present US Army helicopter navigational aids. The activation of the heading lockout submode by the movement of the course set knob is a shortcoming.

97. Several front course instrument landing system (ILS) approaches were flown and evaluated during the program. With the CIS in the ILS mode, the pilot will normally set the course set display to the inbound heading so that the horizontal situation indicator (HSI) will display localizer information in the correct sense. However, while on final approach, if the pilot moves the course set knob (to set missed approach instructions, etc.) the roll bar will be deflected to a new course. With aircraft 77-22716 and the prototype aircraft 77-21652, the roll bar would give valid ILS information after 30 to 45 seconds (without moving the course set knob) which led the pilots to assume the heading lockout submode was being engaged. When the same evaluation was done on aircraft 78-22976 the roll bar would remain off course until the ILS course was reset utilizing the course set knob. Sikorsky personnel could not explain why the roll bar reacted differently in the three aircraft, but stated the roll bar reacted according to design in aircraft 78-22976. In either case the pilot cannot set missed approach instructions prior to the actual missed approach which is contrary to the present procedure used in all other army aircraft. Pilots, due to habit or not fully understanding the system, will set missed approach instructions during the approach thus placing the aircraft off course in close proximity to the ground. It should be noted that the pilot will continue to get valid glide slope information and will continue to reduce his height above ground level even though heading information is incorrect. The erroneous information displayed by the HSI when missed approach instructions are set during the ILS approach significantly degrades safety and is a deficiency.

98. The heading lockout submode significantly increases pilot confusion. Several times during the program both pilots were totally confused due to the contradictory information displayed on the HSI and VSI. The contradictory information was always caused by the engagement of the heading lockout submode and the lack of positive cockpit cues. When working in IMC the pilots should not be required to determine if the information displayed on the HSI or the VSI is correct. Consideration should be given to disabling the heading lockout submode of the CIS.

99. The No. 1 and No. 2 bearing pointers are located on the HSI. In the present configuration the No. 1 bearing pointer can only be used in conjunction with the doppler and the No. 2 bearing pointer can only be used in conjunction with the VOR or automatic direction finder (ADF). Therefore, ADF and VOR information cannot be displayed simultaneously. With this configuration the pilot workload was significantly increased when fixing a VOR/ADF intersection or verifying the position of the aircraft during the interception of the ILS when marked with a locator outer marker. Past aircraft in the Army inventory have had the capability of displaying ADF and VOR information simultaneously. The inability of the HSI of

the CIS to display ADF and VOR navigational information simultaneously is a shortcoming. The HSI configuration of the CIS should be changed so that the No. 1 and No. 2 needle display could be selectable between doppler, VOR, and ADF.

100. The CIS has only a limited capability for automatic interception of VOR radials. If the UH-60A is within 10 to 12 degrees of the selected VOR radial it will automatically display proper roll commands for course interception. However, if the aircraft is greater than 10 to 12 degrees off of the desired VOR radial the pilot must resort to a manual interception procedure which is no improvement over the present systems installed in the Army's utility helicopters. Since the automatic interception feature is so limited, it is virtually useless in its present configuration. The extremely limited capability of the CIS for automatic VOR course interception is a shortcoming. The automatic course interception capability of the CIS should be expanded to provide intercept information for any selected radial.

101. The airspeed hold system is functional whenever airspeed is 60 KIAS or greater and FPS is engaged. Activation of either the four-way trim switch or instantaneous trim synchronizes the airspeed hold and an 18 to 21 second delay is incurred prior to regaining this function. This excessive delay contributed to the moderate pilot compensation required to trim to desired airspeeds. This system degraded the pilot's capabilities in trimming to precise airspeeds, particularly in simulated IMC. Once engaged, the airspeed hold system reduced pilot work load under most conditions; however, the effort required to establish an airspeed is excessive. The excessive delay in engagement of the airspeed hold system following trim actuation is a shortcoming as previously reported (GCT, Nov 1976).

102. The pilot and copilot slip indicators did not show the same indications with the aircraft level. Through the use of an independent lateral accelerometer, it was found that both indications were incorrect. The establishment of a ball-centered indication on the independent lateral accelerometer would result in an approximate 1/8 ball out of trim indication on both ship's trim ball indicators with the balls in opposite directions. The lack of proper alignment of the pilot and copilot slip indicators is a shortcoming and was previously reported (PAE III, Apr 1979).

103. Only one CIS mode select panel is provided. This panel is located on the pilot's side of the cockpit and is not adequately illuminated. The copilot is not provided a CIS mode select advisory panel, therefore, he is unaware of the mode in which the CIS is operating. The lack of a copilot CIS mode select advisory panel is a shortcoming previously reported (PAE IIA, Dec 1978).

104. The CIS is designed to automatically change operational functions at certain points in an instrument task (*i.e.*, VOR level-off, station passage submode, etc.). Pilot cues for some automatic CIS function changes were available; however, they remain inadequate. The lights in the VSI which indicate Go-Around, Decision Height, and Marker Beacon are small, very dim, and poorly located. A positive pilot cueing system to indicate all automatic CIS modes or function changes should be incorporated. The pilot can more effectively fly through the changes and keep himself mentally oriented if he has positive indications that a system command function has changed. An adequate indication might be a light designed to remain ON for about 20 seconds when a command function has changed. Lack of adequate pilot cues for an automatic CIS function change is a shortcoming previously reported, (Project No. 78-01, Mar 1978, ref 23, app A).

105. The level-off function of the NAV mode was designed to be activated by the pilot or copilot LO "bug" radar altimeter setting, whichever was set to the highest altitude. A system based on radar altitude is unacceptable if used over hilly terrain or landing on a plateau surrounded by flat terrain. The system was designed assuming level terrain in the vicinity of the airfield. This assumption could lead to a safety of flight situation whereby the aircraft would not enter the level-off mode if lower terrain is on the approach end of the runway and the pilot failed to monitor his barometric altimeter. The use of radar altimeter settings for the level-off mode could be beneficial in the tactical environment where detailed map reconnaissance is used to document terrain and obstacle elevations. The operator's manual discusses the functioning of the level-off mode briefly; however, the following WARNING should be included in the instrument flight section of the operator's manual:

WARNING

> The use of radar altimeter settings for level-off commands could compromise flight safety if lower terrain is on the approach end of the runway and the pilot fails to monitor the barometric altimeter. Radar altimeter altitude must not be used to determine arrival at minimum descent altitude or decision height. The level-off mode of the CIS should only be used in the tactical environment where detailed map reconnaissance is conducted prior to IMC flight.

106. On numerous occasions various CIS modes engaged or disengaged without pilot inputs. In some cases the mode was engaged and in other cases the mode was not engaged even though the mode select panel indicated it was. The majority of the uncommanded "engagements" usually occurred during engine starts or during flight in light turbulence. However, several of the uncommanded "engagements" appeared to be random in nature. The uncommanded "engagements" of CIS modes is a shortcoming.

107. The CIS has the potential to be a very effective all-weather navigation and aircraft control tool; however, in its present state it is well below its potential. The inability to preselect an airspeed, a rate-of-climb or descent, and a barometric altitude for cruise level-off in addition to the shortcomings and deficiency listed above severely degrades the usefulness of the CIS. The CIS usefulness is further degraded by the complexity of the VSI (two control knobs, three advisory lights, four caution flags, seven types of indicators in addition to the attitude indicator and normal markings).

In-flight Engine Starts

108. The engine restart characteristics were evaluated in-flight on UH-60A S/N 77-22717. These tests were conducted in both the IR normal utility configuration and the normal utility configuration including the XM-130 chaff dispenser. Both the auxiliary power unit (APU) and engine crossbleed start methods were evaluated at pressure altitudes of 20,000 feet and below, and at airspeeds between 40 and 90 KIAS. The effects of sideslip resulting from a one ball width displacement on the pilot's turn and slip indicator were evaluated at 70 KIAS. A summary of the successful engine starts using the procedures outlined in the

operator's manual is presented in table 8. Representative time histories of a successful in-flight start and an aborted start are shown in figures 111 and 112, appendix E, respectively.

Table 8. Successful In-flight Engine Starts

Airspeed (KIAS)	Method	Pressure Altitude[1] (ft)	OAT (°C)
90	CB[2]	17,000	-9.0
	APU	17,000	-8.5
70	CB	18,000	-16.0
	APU	18,000	-16.0
40	CB	16,000	-8.0
	APU	17,000	-16.0

NOTES:

[1] Highest pressure altitude both engines started
[2] CB = Crossbleed

109. A total of 39 start attempts were performed with 25 successful starts. Six of the 14 aborted start attempts were aborted by manually disengaging the starter at 60 seconds after the start cycle was initiated. This procedure was based on paragraph 8-26.8.g(2) of the operator's manual which states the maximum time from N_g indication to idle should not be over 45 seconds at outside air temperature (OAT) above -20° C. Below -20° C and at pressure altitudes above 10,000 feet allowable start time is 60 seconds. After the starter was manually disengaged, the N_g decreased with a corresponding TGT increase and the start was aborted at 850° C.

110. The time from starter engagement to maximum N_g speed with PCL off and the TGT/time gradient after the PCL was placed in the IDLE detent were essentially the same regardless of aircraft configuration. Neither the IR suppressors nor airflow change due to sideslip appeared to have a significant effect on the engine start characteristics at the conditions tested.

111. Paragraph 3.7.5.8.1.C of the PIDS requires engine starts during flight up to the service ceiling of the aircraft (20,000 feet pressure altitude). The UH-60A failed to meet this requirement by 2000 feet in that 18,000 feet pressure altitude was the highest altitude at which any successful in-flight engine starts were accomplished. Successful starts were always achieved between 40 and 90 KIAS at or below 16,000 feet pressure altitude regardless of the start method used. Seventy KIAS appeared to be the best airspeed to perform the engine starts since starts were

accomplished at higher pressure altitudes (18,000 feet). In addition, at 70 KIAS, a lower rate of descent is attained and more time is available to perform an in-flight engine start.

Aircraft Systems Failures

Automatic Flight Control System Failure:

112. A limited evaluation was conducted on the analog (No. 1) SAS. The evaluation consisted of hardover tests in the lateral, longitudinal and directional axis at 100 and 123 KIAS. SAS hardovers in all three axes resulted in very mild attitude changes and the highest rate observed was approximately 5 deg/sec in roll. The results of the analog SAS hardover evaluation were satisfactory.

Pitch Bias Actuator Failures:

113. During the A&FC evaluation, random PBA failures resulted in numerous flights being delayed or aborted. Approximately 20 percent of the time the PBA failures could be corrected by resetting the AFCS "POWER ON RESET" buttons. The only indication of a PBA failure was the illumination of the PBA caution light. Qualitatively, the pilots were unable to determine the status of the PBA without the caution light. The PBA can fail in any position during flight. If the failure occurs at the extreme forward position, 1.5 inches of aft cyclic control authority will be lost and at the extreme aft position, 1.5 inches of forward cyclic control authority will be lost. As a result of the numerous PBA failures and the apparent lack of pilot cues, such as attitude changes, a short evaluation was performed to determine if the PBA enhanced the static stability of the aircraft. Three tests were conducted with the PBA locked in the centered position: control positions in trimmed forward flight from 46 to 151 KCAS (fig. 90, app E), static longitudinal stability during level flight at 87, 123, and 153 KCAS (figs. 91 and 92), and maneuvering stability at 131 KCAS (power for level flight) (fig. 94). Additionally, figures 90, 91, and 93 present comparisons with the PBA operational. The data with the PBA operational for static longitudinal stability was taken from reference 13, appendix A. The results of the comparison show there were no significant changes in control positions or static longitudinal stability when compared with the PBA locked in the centered position. The maneuvering stability comparison shows no change in longitudinal cyclic control position; however, the lateral cyclic control position did show a shallower gradient in left turns. This difference was not perceptable to the pilot during the flight. Due to the high failure rates, the possibility of losing 1.5 inches of longitudinal control authority, and no apparent benefit of the PBA, the pitch bias actuator should be locked in the centered position and phased out of the UH-60A fleet as supplies are diminished.

VIBRATION CHARACTERISTICS

114. Vibration characteristics were qualitatively evaluated throughout the test program and quantitatively evaluated at the conditions listed in table 1 during the level flight performance and maneuvering stability tests. Vibration sensors were installed as indicated in appendix C. Only 4/rev vibration data test results are presented in figures 113 through 124, appendix E, as the other harmonics were not significant. Table 1, appendix B, gives a brief description of the vibration absorbers used in the A&FC along with tuning weights.

115. Representative level flight vibration data for the pilot station are presented in figures 113 through 115, appendix E. In level flight, the vibration characteristics near mission and alternate gross weights were similar at the pilot's station for airspeeds up to 80 KCAS. As shown in figure 115, the pilot seat vertical and longitudinal acceleration at the 4/rev frequency (17.2 Hz) increased as airspeed was increased. The vibration levels at the higher airspeeds (above 125 KCAS) were moderate and could affect the aircrew over a long period (VRS 5). Figure 113 shows a lower vibration level in the vertical axis and higher lateral acceleration above 110 KCAS at essentially the same aircraft configuration but warmer temperature. This change in vibration characteristic was noted by the pilots. During the test program, many modifications were made to the test aircraft by Sikorsky Aircraft personnel in their effort to update the aircraft to a production model. These changes included such items as spindle assembly, dampers and damper bolts, etc., which may cause a variability of vibration characteristics. Because of these changes, the aircraft exhibited different vibration characteristics. Therefore, throughout the test program, the vibrations were qualitatively assessed VRS ratings ranging from 2 to 7. The vibration characteristics did not meet the requirement of paragraph 3.2.1.1.3.1.4 of the PIDS. Future tests should be conducted on a current production UH-60A in order to obtain vibration data that would be more representative of the aircraft.

116. Vibration levels at the cg of the aircraft during level flight tests are presented in figures 116 through 118, appendix E. At the heavy weight condition as shown in figure 118, the vertical acceleration exceeded 0.2 g at 55 KCAS and fails to meet the requirements of paragraph 3.2.1.1.3.1.4 of the PIDS. The lateral and longitudinal accelerations were below 0.05 g at all airspeeds. At 16,480 pounds (approx primary mission gross weight), below 56 KCAS and above 120 KCAS the vertical acceleration exceeded 0.1 g with a peak of 0.17 g at 148 KCAS (fig. 117). The lateral and longitudinal accelerations were essentially below 0.05 g at all airspeeds. The data shown in figure 116 was obtained in a warmer environment than that shown in figure 117. In both cases the vertical acceleration was essentially below 0.1 g throughout the airspeed range and the lateral and longitudinal acceleration were below 0.05 g.

117. The vibration characteristics of the pilot's and copilot's instrument panel were qualitatively evaluated throughout the test program. The pilots were able to read the instruments and no blurring was ever experienced. Because of the location of the accelerometers, as explained in the test techniques and data analysis section (app D), no quantitative panel data was included in this report.

118. Level flight performance tests were conducted at a rotor speed of 245 rpm (95 percent minimum power on rotor speed) and the 4/rev (16.3 Hz) vibration characteristics at this rotor speed are shown in figure 119 through 121, appendix E. Comparing the lower rotor speed data at heavy and primary mission gross weights to the higher rotor speed at similar conditions, show essentially no difference in vibration level at the pilot's station. The longitudinal vibration showed no difference in amplitude. The lateral and longitudinal vibration level at the primary mission gross weight and 245 rpm rotor speed were similar at the cg location. However, the vertical acceleration, showed an increase in amplitude below 60 KCAS and above 120 KCAS. The vertical vibration levels at the aircraft cg, minimum power-on rotor speed, airspeeds below 50 KCAS at both gross weights and above 138 KCAS at the primary mission gross weight, exceeds 0.15 g and does not meet the requirements of paragraph 3.2.1.1.3.1.4 of the PIDS.

119. Vibration test results near primary mission gross weight and aft cg during maneuvering flight are presented in figures 123 and 124, appendix E. At 118 KCAS, the highest 4/rev vibration level was 0.24 g laterally at the pilot seat, which occurred at a bank angle of 45 degrees or 1.4 g normal acceleration. At a 30 degree, ball centered, right bank the vibration level was slight (VRS 3). Deviating from ball center to a 3 degree right sideslip increased the vibration to a moderate level (VRS 4). A six degree left sideslip caused an additional increase (VRS 5). In a 30 degree left bank and a six degree left sideslip the vibration level was greater than that experienced during right bank (VRS 6), and airspeed was more difficult to control. During the 45 degree bank, the aircraft had an inherent five degree left sideslip while, holding ball centered flight. Decreasing the sideslip to two degrees left while in a 45 degree left bank decreased the vibration level, while increasing the sideslip to 8 degrees left increased the vibration level (VRS 5). The 60 degree left and right maneuvering stability flight was difficult to perform because the airspeed fluctuated ±10 KIAS and pilot workload increased significantly (HQRS 6). Future tests should include maneuvering stability tests at the alternate design gross weight to determine if the vibration levels are as high as those reported during the GCT.

120. The vibrations were excessive during descents and translations from forward flight to hover, as well as in those areas previously mentioned. The overall vibrations have not been improved from those previously noted in other reports. Other harmonics (1, 3, and 8/rev), were evaluated and are satisfactory. The excessive vibrations significantly increased pilot work load during certain maneuvers, and are a shortcoming.

HUMAN FACTORS

General

121. The cockpit layout, which includes switch function design and position, instrument position, available cues, and procedures were evaluated during day, night, and simulated IMC flights.

Cockpit Evaluation

122. The light intensity of the caution panel segment legends, when illuminated, was insufficient when exposed to direct sunlight. This made it difficult in determining which caution segments were illuminated and would delay the pilot's reaction to emergency conditions. The lack of readily discernible caution panel segment lights in bright sunlight is a shortcoming previously reported (PAE IIA, Oct 1978).

123. The pilots were unable to detect illuminated advisory segments in direct sunlight; therefore, were unaware of relevant changes in aircraft systems operation (*i.e.*, back-up pump ON). By not being aware of all advisory information it would be possible for the pilot to incorrectly identify various emergencies. The lack of readily detectable advisory lights in direct sunlight is a shortcoming previously reported (PAE IIA, Oct 1978).

124. The function select switches (SAS 1, SAS 2, TRIM and stabilator AUTO control) on the AFCS switch panel were too dim to allow the pilot to read them under normal daylight conditions. The inability of the pilot to readily determine

AFCS function select switch position delays the pilot's recognition of a degraded flight control system. The lack of readily detectable switch position on the AFCS switch panel is a shortcoming previously reported (PAE IIA, Oct 1978).

125. The production UH-60A was equipped with a C-6533/ARC intercommunication control panel (ICP). Due to the ICP configuration which has no intercom ON/OFF switch, the pilot was forced to monitor the aircraft intercom communications at all times. This leads to extreme communication problems when one pilot must converse with the troop commander or crew chief and the other pilot must converse with air traffic control or ground elements. The lack of an ON/OFF switch on the aircraft's intercom system is a shortcoming previously reported (PAE IIIA, Aug 1979).

126. The UH-60A was equipped with a transponder set (AN/APX-100(V)1) to provide automatic radar identification. Due to the poor design of the Mode 2 numeral cover (photo 2) on the control panel (RT-1296/APX-100(V) it was extremely difficult to set transponder codes. The code windows which are recessed below the Mode 2 numeral cover collected dust which made identification of transponder codes extremely difficult. To adequately clean the code windows, the Mode 2 numeral cover must be removed and the windows cleaned. In dusty conditions this process must be done on a daily basis. The difficulty in setting codes in the transponder due to dust collection on the code windows is a shortcoming.

127. In order to set a transponder code, a code selector button must be depressed to advance the code number, one digit each time the button is depressed. The design does not allow the pilot to rotate the code number backwards; for example, from 7 back to 6. Instead, if the digit showing is 7, in order to select the digit 6, he must depress the code selector button 7 times. This can require the pilot to depress each of 4 Mode 3/A code selector buttons a total of 7 times which requires excessive time and distraction from the external flight environment. The poor design and operation for the code select buttons on the RT-1296/APX-100(V) is a shortcoming.

128. During rapid decelerations in the UH-60A a very nose high attitude was required. As a result of the nose high attitude, the forward field of view was significantly reduced by the instrument panel glare shield. The forward field of view could be improved somewhat by utilizing the "chin bubble;" however, due to the limited size of the "chin bubble," the forward field of view was still very restricted. The restricted forward field of view during nose high attitude is a shortcoming.

129. The UH-60A design incorporates a crossbar above the pilot's windows which doubles as a handhold. The crossbar is approximately 5 inches wide and runs the full width of the cockpit. During any maneuver requiring a nose low attitude such as level acceleration or high accelerative type takeoffs, the crossbar significantly obstructs the pilot's forward field of view. Due to the significant loss of forward field of view the pilot was required to look out the overhead cockpit window during nose low attitude maneuvers and would still only have marginally forward field of view. The loss of forward field of view during nose low maneuvers is a shortcoming.

130. Quartering rearward field of view was severely restricted by a side structured bulkhead just to the rear of the pilot's and copilot's seats. When hovering rearward, this loss of quartering rearward field of view forces the pilot to rely on the crew

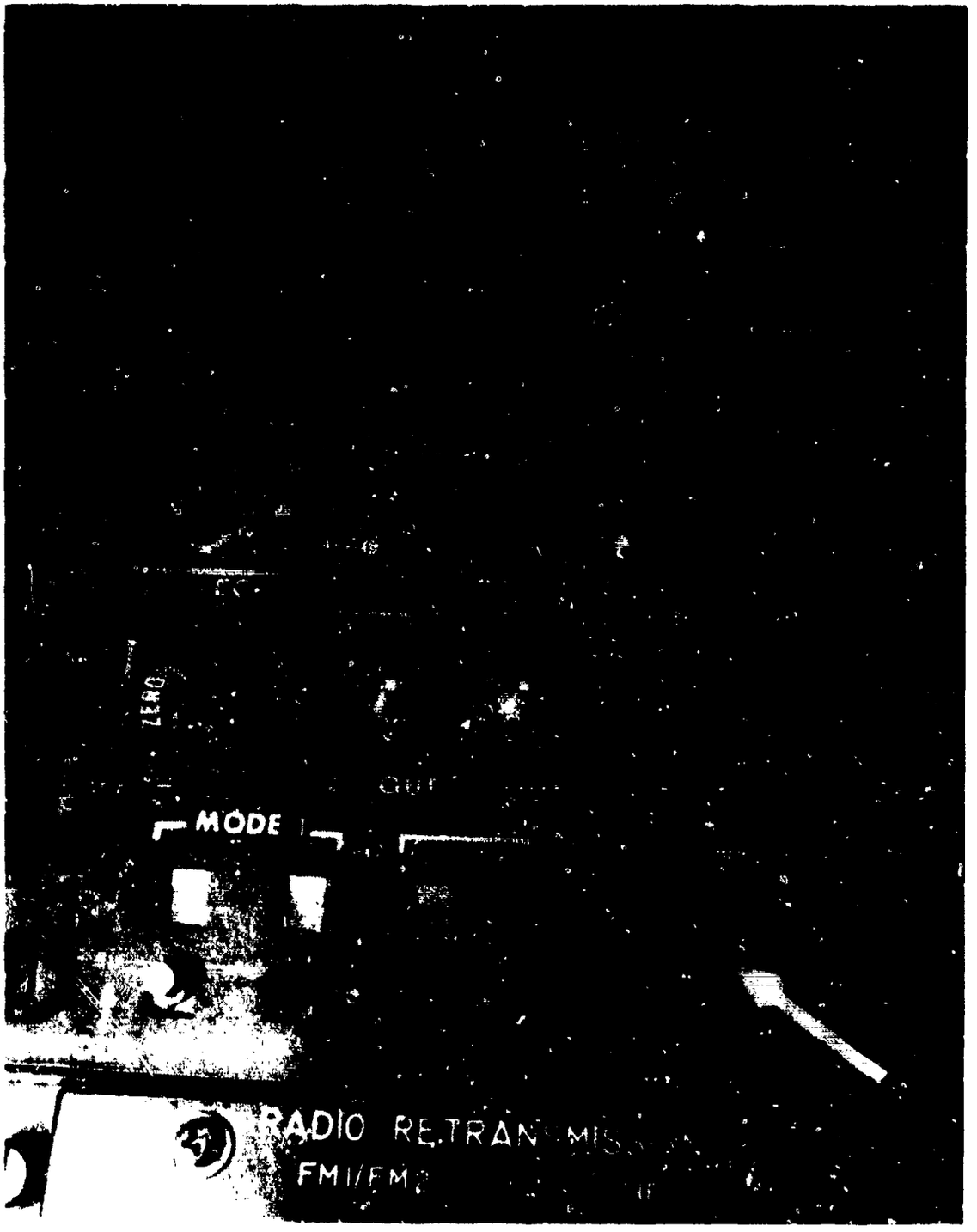


Photo 2. Transponder Control Panel RT-1296;APX-100(V)

chief's verbal instructions; whereas, in a UH-1, the pilot is not as dependent on the crew chief due to a quartering rearward field of view. The severely restricted quartering rearward field of view is a shortcoming. A similar shortcoming was reported during GCT, November 1976.

131. The UH-60A was equipped with a foam rubber extension on the cockpit glare shield. The pilot's and copilot's warning lights were partially obscured by this glare shield extension. Partial obscuration of the warning lights by the glare shield extension is a shortcoming previously reported (PAE IIIA, Aug 1979).

132. The UH-60A caution/advisory panel is equipped with an oil pressure caution light for each engine. When operating at low gas producer speeds such as ground taxiing, the oil pressure lights illuminate and when cross-checked with oil pressure gauges the low oil pressure is verified. In discussing the problem with Sikorsky maintenance personnel, it was determined the indicators and gauges are correct and the engines are operating at low oil pressure. It was not determined if any damage would be done by operating at low oil pressures for extended time periods (*i.e.*, quick reaction type standby). Low engine oil pressure indications by oil pressure gauges and caution lights at low gas producer speed is a shortcoming previously reported (PAE IIIA, Aug 1979). The low oil pressure indications at low gas producer speeds should be investigated and, if determined to be nondamaging, the cautionary zone on the engine oil pressure gauges and the activation mechanism for the engine oil pressure caution lights should be lowered.

133. The pilot's and copilot's seats have a manual adjustment of approximately 5 inches vertically and longitudinally. The seats are easily adjusted; however, for almost all pilots (regardless of physical stature), the seats do not adjust sufficiently aft for a comfortable pilot position. The lack of adequate aft seat adjustment is a shortcoming previously reported (GCT, Nov 1976).

134. The pilots' seats are designed with armor plate side panels which extend forward of the seat back vertical plane. The seat back side panels restricted the 95 percentile pilot's arm movement when making collective adjustments during any flight maneuvers. This caused the pilot to lean left or right from a vertically seated position to prevent his elbow from hitting the seat side panel when the collective was raised. The design of the pilots' seat back side panels restricts arm movement when making collective adjustments and is a shortcoming.

135. The alternating current (AC) and direct current (DC) primary bus circuit breakers are located on an overhead panel located aft of the pilot's and copilot's heads. It is impossible to see the panels without considerable head rotation (aft and up), which can induce vertigo and/or spatial disorientation. In the event of emergency action requiring a circuit breaker to be pulled, the pilot and/or copilot must disassociate his attention from the instrument panel and the flight path to operate the circuit breakers. The inaccessibility of the AC and DC overhead circuit breaker panels is a shortcoming previously reported (GCT, Nov 1976).

136. The UH-60A has a takeoff and landing checklist permanently installed on the instrument panel. At the present time the checklist is engraved on a piece of acrylic and cannot be changed. Since the checklist cannot be changed, it does not reflect the current takeoff and landing checklists, thus nullifying any practical usage. The cockpit installed checklist design should be modified so that it can be changed to reflect current procedures.

Night Evaluation

137. The internal cockpit lighting and lighting controls were evaluated on the ground and in flight. The secondary lights/cockpit floodlights were mounted between the pilot and copilot and were controlled by one three-position switch (Red, Remote, and White), and two rotary dimmer controls. The prototype UH-60A utilized four controls for the same functions. Even though one control has been elliminated, the secondary lights/cockpit floodlight controls are unnecessarily complicated. The functions of these multiple controls should be consolidated. The use of multiple secondary lights/cockpit floodlight controls is a shortcoming previously reported (GCT, Nov 1976). Consideration should be given to removing the Red/Remote/White switch on the secondary lights/cockpit floodlight and incorporating the secondary and cockpit floodlight reostats on the battery utility bus.

138. None of the circuit breaker panels installed on the UH-60A have provisions for night lighting. Several emergency actions require the pilots to pull circuit breakers. Due to the large number of circuit breakers in the UH-60A, it was very difficult to find and identify the proper circuit breaker under daylight conditions, much less under night lighting. The lack of night lighting for the circuit breaker panels is a shortcoming.

139. The engine control quadrant was not equipped with a night lighting capability. The lack of a quadrant lighting system delayed crew recognition of engine control lever and fuel selector positions. The lack of an engine control quadrant lighting system is a shortcoming previously reported (PAE IIA, Oct 1978).

Noise Evaluation

140. The noise level in and around the UH-60A was qualitatively evaluated during all phases of the A&FC tests. The noise level in the cockpit and the forward portion of the cabin was objectionably high. With the bleed air heater on, or vent blower on, or either cockpit door window open the noise level at both stations significantly increased. The USAAEFA aircraft did not have soundproofing due to the unique instrumentation requirements; however, other UH-60A's were flown with a full compliment of soundproofing and no significant reduction in noise level was noted. The objectionably high cockpit and cabin noise level is still a shortcoming previously reported during GCT, Nov 1976.

141. The APU system provides pneumatic power for main engine starting, cabin heating, and electrical power for ground and emergency in-flight electrical operations. Maintenance personnel and flight crew experienced extreme discomfort when working around the aircraft with the APU running. Personnel were virtually unable to verbally communicate unless on the intercom. An external power cart was utilized more frequently throughout the A&FC than the APU for ground systems checkout because of the noise discomfort. The APU, however, can be expected to be used more frequently in an operational environment. The high noise level of the APU is still a shortcoming previously reported during GCT, Nov 1976.

RELIABILITY, AVAILABILITY, AND MAINTAINABILITY

142. During the course of the A&FC, many major components on the UH-60A had to be replaced. Table 9 is a partial list of items replaced on aircraft 77-22716 during the A&FC. This list does not include any part changes due to modification 99, time change items or part changes resulting from special inspections. The list only includes parts that were replaced due to "failure" of the part or the inability of the part to pass scheduled inspections. It should also be pointed out that only a total of 218.8 flight hours were flown on aircraft 77-22716. The reliability, availability and maintainability (RAM) data on the UH-60A should be reviewed and closely monitored.

143. The UH-60A was equipped with rubber splash guards mounted on the fuselage designed to seal the oleo struts in the fuselage. These splash guards deteriorate very quickly and must be cut away from the aircraft so that they will not separate in-flight. The quick deterioration of the oleo strut splash guards is a shortcoming and was previously reported in PAE IIIA (Aug 1979).

144. The avionics compartment door did not have any upper stops. In the event that the avionics compartment door is open and unattended during windy conditions, the aircraft will be damaged with upward movement of the door which will impact on the center windshield and windshield wipers. The lack of upper stops on the avionics compartment door is a shortcoming previously reported (PAE IIIA, Aug 1979).

145. With the IR suppressor installed, the UH-60A fuselage skin was damaged due to excessive heat resulting in "hot spots." The damaged areas were located in the vicinity of the cooling air inlet. Prior to the discovery of the damage the aircraft was flown approximately 1.2 hours at high power tethered hover. Prior to the tethered hover flight, the aircraft had been flown for several hours with the IR suppressors installed and no damage was found. It should be noted that toxic fumes associated with burning Kevlar were very prevalent in the vicinity of the damaged areas for several hours. The design of the IR suppressor for the UH-60A does not allow the suppressor to be used during continuous high powered hover without damaging the aircraft and is a shortcoming. The significance of heat damage during high power hover with the IR subsystem installed should be further evaluated.

146. The UH-60A main transmission is equipped with a dipstick for checking the oil level. During the A&FC tests it was found that the dipstick would not give reliable indications of fluid level. The oil was checked prior to the first flight and prior to turning the main rotor and would indicate a low oil level. Reinserting the dipstick and rechecking the oil level, it was found the level would normally increase one quart. The oil level indication could be further increased by rotating the main rotor approximately one revolution. The lack of a reliable main transmission oil level indication is a shortcoming.

147. The UH-60A was plagued with chip lights during the A&FC tests. Approximately 40 percent of the occurrences required maintenance personnel to remove and clean the chip detectors. None of the occurrences required any component changes. It was not unusual to have several occurrences on different modules during the same flight. Due to the large number of false chip lights caused by insignificant particles (fuzz, carbon particles, etc) missions will be delayed or aborted unnecessarily. The excessive frequency of illumination of chip lights due to insignificant particles is a shortcoming and will significantly impact mission availability.

Table 9. Partial List of Major Component Changes Due to Failure or Unserviceability on Aircraft 77-22716 During the A&FC Evaluation

Number Replaced	Item	Part Number
2	Turbine Engine	7000-10100-011
1	Auxiliary Power Unit (APU)	70303-03100-041
1	Main Rotor Blade	70150-09100-043
1	Main Rotor Spindle	70102-08100-051
1	Pitch Bias Actuator	70400-02200-107
2	Generator	70550-02031-111
3	Hydraulic Module	70652-02110-106
1	Roll Trim Actuator	70400-02260-111
1	Tail Rotor Gear Box	70358-06600-042
1	Generator Control Unit	70550-02007-013
1	APU Accumulator	70651-03201-102
3	Blade Inspection Method Indicator (BIM)	26115-20520-001
1	Horizontal Situation Indicator	70450-01040-113
1	Tail Wheel Locking Actuator	70250-13006-102
1	External Power Relay	M24021-2
1	Starter Motor	70306-10000-103
2	Hydromechanical Unit	6038T62P04
2	Roll Rate Gyros	TRU-3A/A
1	Deice Valve	5044T08P03
2	Forward Engine Mount Elastomeric Bearing	70361-08001-101
5	Fuse Limiter	MS28937-60

148. Several anomalies or failures of various systems occurred during the A&FC. The majority of anomalies were in the CIS or AFCS subsystems. In general, most of the failure indications were false; however, some were actual failures and could usually be corrected by resetting various controls within the cockpit. It was felt that the vast majority of the anomalies were caused by EMI. The more common anomalies are listed below:

a. Stabilator failure during engine shutdown

b. Random trim and or FPS failures

c. Various CIS modes engaging during engine start

d. Lateral "ACCL" failure advisory during engine start

e. Lateral "ACCL" failure advisory during takeoffs

f. Various computer maintenance latches latched randomly

g. Random PBA failures.

Maintenance personnel normally are warned of impending maintenance problems by various anomalies which occur on Army aircraft. The anomalies on the UH-60A are so random and frequent they are ignored and of little use to maintenance personnel. The various anomalies which occur in the AFCS and CIS subsystems is a shortcoming.

149. The AFCS computer is equipped with maintenance latches which will automatically latch in the event of a computer malfunction. Throughout the A&FC the maintenance latches latched (indicating a malfunction) even though there was no malfunction. The latching was random in nature and, excluding the "FAN FAIL" latch, all latches did latch during the A&FC. One to five maintenance indicators would latch during a flight. The random latching of the AFCS computer maintenance latches is a shortcoming and was previously reported (PAE IIA, Dec 1978).

150. The master caution light would illuminate after being reset during dual SAS OFF flight. Further investigation revealed that the actuation of the pilot's or copilot's cyclic trim release would illuminate the master caution light. The illumination of the master caution light by the pilot's cyclic trim release during dual SAS OFF flight is a shortcoming previously reported (PAE III, Apr 1979).

151. During shutdown of the APU, the external power source would not automatically accept the aircraft electrical loads. Electrical power was restored to the aircraft when the external power switch was cycled to RESET and then returned to the ON position. The failure of the external power source to immediately assume the aircraft electrical load upon APU shutdown is a shortcoming previously reported (PAE III, Apr 1979).

152. Due to the design of the tail rotor gear box cowling, it was very difficult to ascertain the oil level of the gear box. On several occasions, due to the ambient light condition, the oil level could only be determined by climbing up the vertical fin

which increased turn-around time. The inability to determine the oil level in the tail rotor gear box from the ground is a shortcoming previously reported (PAE III, Apr 1979).

153. During preflight inspections of the cabin top areas, it was noted that the designated step areas over the crew chief and gunner windows appeared to contain internal separation. When stepping on these areas, the step surface would yield under normal foot pressure and internal scraping sounds were detected. These areas are used repeatedly for preflight inspections and maintenance work, and their apparent weakness indicates possible early failure. The internal separation of the designated cabin top step areas is a shortcoming prevously reported (PAE III, Apr 1979).

154. The cockpit doors are secured in the closed position by lugs extending into a latch when the door release handle is in the locked position. If the door is closed with the door handle locked, the securing lugs knock the latch out of position, precluding subsequent securing of the doors without mechanically readjusting the latch. Under these conditions, damage to the lugs and/or the latch is probable. The inability to close the cockpit doors with the release handle in the locked position without causing damage to the door locking mechanism is a shortcoming previously reported (GCT, Nov 1976).

155. The APU compartment access doors, located on the top fuselage aft of the main rotor mast, are designed such that in order to secure both doors the left door is closed after and overlaps the right door. The left door contains the two access door latches. This design allows the left door to be closed and latched first which does not secure the right door. If the right APU access door is not secure it may be blown open in flight and contact the main rotor blades. The inability to easily determine the security of the right APU compartment access door is a shortcoming. The following CAUTION should be included in chapter 8 of the operator's manual:

CAUTION

Insure both APU comparment access doors are secured prior to operating the main rotor.

A brightly colored strip should be painted on the overlapped portion of the right APU compartment access door to reduce the possibility of not securing right door prior to operating the main rotor.

AIRSPEED CALIBRATION

156. Airspeed calibration tests were conducted to determine the position error of the UH-60A airspeed system. The aircraft's pitot-static system was calibrated over a ground speed course in level flight, and by use of a calibrated trailing bomb (finned pitot-static system) in climbs and descents. Results of these tests are presented in figures 125 and 126, appendix E.

157. In level flight, airspeed position error varied from -12 knots at 30 KIAS to nearly zero between 130 and 140 KIAS to -1 knot at 160 KIAS. In autorotation, the ship's system position error was nonlinear and varied from -27 knots at 15 KIAS to zero at 67 KIAS to +5 knots at 129 KIAS. The large errors and variation in position error below 40 KIAS in autorotation, and large errors in level flight below 40 KIAS

resulted in ship's airspeed system being unuseable below this point. In climbing flight at MCP, the ship's system position error was nonlinear and varied from -1 knot at 47 KIAS to a maximum of -8 knots at 87 KIAS to -5 knots at 120 KIAS. Airspeed indicator fluctuations of ±5 knots occurred in climbing flight between 60 and 70 KIAS. Below 70 KIAS in climbs, airspeed indications appear to be affected by power setting, *i.e.*, increasing collective increases error (positive increase in correction to be added). The large variable airspeed position error in various flight regimes is a shortcoming.

MISCELLANEOUS

158. The following deficiencies and shortcomings were not re-evaluated during the A&FC. Some of these items may have been corrected due to design or procedural changes.

a. Unexplained shifting of the lateral rigging (PAE, Apr 1979)

b. The inadequate pilot warning cues of partial power engine malfunctions (PAE IIIA, Aug 1980)

c. The lack of a reliable FM station passage indication (PAE IIA, Dec 1978)

d. Significant power loss associated with operation of the engine anti-ice and cockpit heater system (GCT, Nov 1976)

e. Excessive engine/rotor speed transients which occurred following large magnitude collective application to the opposite extremes of the power demand schedule (GCT, Nov 1976)

f. At a reference power turbine speed of 20,480 rpm (98 pct or 253 rpm main rotor speed) unacceptably large transient droop (Project No. 77-58, Dec 1977)

g. Unstable longitudinal control force stability at representative NOE airspeeds (40 KCAS and slower) (GCT, Nov 1976)

h. Inability to start the YT700-GE-700 engines at high altitude landing sites without first achieving a successful vapor vent (GCT, Nov 1976)

i. Difficulty in achieving a successful fuel system vapor vent at high altitude (GCT, Nov 1976).

CONCLUSIONS

GENERAL

159. Based on the A&FC evaluation of the production UH-60A helicopter, the following conclusions were reached:

a. The UH-60A is a marked improvement over previous utility helicopters; however, many problems exist relative to airworthiness and flight characteristics

b. Some of the handling qualities tests were completed for inclusion in the operator's manual; however, the handling qualities objective of this program was not met due to limitations imposed on the program.

c. The performance of the UH-60A has been improved over that of the YUH-60A due to the lower primary mission gross weight, the reduced power required and the increase in power available.

d. Seven deficiencies, 64 shortcomings, 10 items that did not comply with the PIDS, and one possible PIDS noncompliance were noted.

ENHANCING CHARACTERISTICS

160. The following enhancing characteristics were identified:

a. The excellent gust response of the UH-60A with the AFCS engaged (para 47)

b. The effective torque-matching capability of the T700-GE-700 engines (para 82)

c. The capability of simultaneous dual engine starts (para 54).

DEFICIENCIES

161. The following deficiencies (in order of their importance) were identified:

*a. The nose down pitching moment experienced during takeoff and climbout (para 63)

b. The inability to control rotor speed in engine ECU lockout with the 404T52G01 HMU (para 85)

*c. The unreliability of the AFCS heading hold and subsequent large yaw excurison (para 67)

d. The design of the idle detent cam on the engine control quadrant which allows inadvertent engine shutdowns (para 87)

*Reported during previous evaluation

*e. The inability to achieve full aft longitudinal cyclic control and the restriction of lateral cyclic control (para 37)

*f. The uncommanded directional control input in trimmed forward flight (para 35)

g. The erroneous information displayed by the HSI when missed approach instructions are set during an ILS approach (para 79).

SHORTCOMINGS

162. The following shortcomings (in order of their importance) were identified:

*a. The large lateral control displacement required to maintain the desired ground track during takeoffs and landings (para 69)

*b. The lack of readily discernable caution panel segment lights in bright sunlight (para 122)

*c. The lack of readily detectable advisory lights in direct sunlight (para 123)

*d. The large lateral control jump (para 34)

*e. The lack of an ON/OFF switch on the aircraft's intercom system (para 125)

*f. The lack of adequate aft seat adjustment (para 133)

*g. The poor engine droop characteristics (para 86)

*h. The failure of the trim system during level accelerative takeoffs (para 70)

i. The excessive vibrations during certain maneuvers (para 120)

j. The pitch oscillation in low speed flight with high rates of climb (para 44)

*k. The self-excited divergent mechanical instability during ground operation (para 60)

*l. The aircraft pitch oscillation (para 59)

m. The lateral shuffle in low speed flight at relative wing azimuths between 285 and 345 degrees at speeds of 15 to 25 knots (para 81)

*n. The lack of readily detectable switch position on the AFCS switch panel (para 123)

o. The sensitivity of rotor speed to collective movement (para 28)

*Reported during previous evaluation

p. The inability to use the No. 1 FM radio due to EMI problems (para 92)

*q. The lack of a mechanical forward cyclic control stop (para 33)

r. The limited load factor envelope at high gross weights (para 49)

s. The activitation of the heading lockout submode by the movement of the course set knob (para 96)

*t. The lack of an engine control quadrant lighting system (para 139)

u. The lack of night lighting for the circuit breaker panels (para 138)

*v. The severely restricted quartering rearward field of view (para 130)

w. The design of the IR suppressor for the UH-60A does not allow the suppressor to be used during continuous high powered hover without damaging the aircraft (para 145)

*x. Partial obscuration of the warning lights by the glare shield extension (para 131)

*y. The erratic airspeed indications during aircraft acceleration below 60 KIAS (para 66)

z. The excessive frequency of illumination of chip lights (para 147)

aa. The inability of the DPLR NAV submode of the CIS to display distances and ground speed in units of nautical miles (para 93)

bb. The poor design and operation for the code select buttons on the RT-1296/APX-100(V) (para 127)

cc. The difficulty in setting codes in the transporder due to dust collection on the code windows (para 126)

dd. The lack of an indication of the commanded tail wheel locking pin position (para 58)

*ee. The large N_F and N_R overshoots that occur with the release of the engine overspeed test button (para 56)

*ff. The lack of upper stops on the avionics compartment door (para 144)

gg. The inability to easily determine the security of the APU compartment access door (para 155)

hh. The large variable airspeed position error in various flight regimes (para 157)

*Reported during previous evaluation

*ii. The random latching of the AFCS computer maintenance latches (para 149)

jj. The uncommanded "engagements" of the CIS modes (para 106)

*kk. The difficulty in unlocking and locking the tail wheel (para 57)

ll. The extremely limited capability in the CIS for automatic VOR course interception (para 100)

mm. The inability of the HSI of the CIS to display ADF and VOR navigational information simultaneously (para 99)

nn. The restricted forward field of view during nose high attitude (para 128)

oo. The loss of forward field of view during nose low maneuvers (para 129)

pp. The inability of the pilot to utilize the VSI (roll bar) of the CIS during holding patterns at a VOR (para 95)

*qq. Lack of adequate pilot cues for an automatic CIS function change (para 104)

*rr. The lack of a copilot CIS mode select advisory panel (para 103)

*ss. The jarring pitch response to a cyclic release following a longitudinal cyclic displacement against trim (para 52)

*tt. The internal separation of the designated cabin top step areas (para 153)

*uu. The excessive delay in engagement of the airspeed hold system following trim actuation (para 101)

vv. The design of the pilots' seat back side panels (para 134)

*ww. The inaccessibility of the AC and DC overhead circuit breaker panels (para 135)

*xx. The requirement for the pilot to consciously reposition his feet to engage heading hold (below 60 KIAS) (para 68)

*yy. The decrease in roll rate following the initial peak (para 50)

*zz. The inability to close the cockpit doors with the release handle in the locked position without causing damage to the door locking mechanism (para 154)

*aaa. The inability to determine the oil level in the tail rotor gear box from the ground (para 152)

bbb. The lack of a reliable main transmission oil level indication (para 146)

*Reported during previous evaluation

*ccc. The objectionably high cockpit and cabin noise level (para 140)

*ddd. The high noise level of the APU (para 141)

eee. The erroneous TRIM ACT indication on the maintenance panel following the hydraulic leak test system check (para 55)

*fff. The illumination of the master caution light by the pilot's cyclic trim release during dual SAS OFF flight (para 150)

*ggg. Low oil pressure indications by oil pressure gauges and caution lights at low gas producer speed (para 132)

*hhh. The use of multiple secondary lights/cockpit floodlight controls (para 137)

*iii. The failure of the external power source to immediately assume the aircraft electrical load upon APU shutdown (para 151)

*jjj. The lack of proper alignment of the pilot and copilot slip indicators (para 102)

kkk. The various anomalies which occur in the AFCS and CIS subsystems (para 148)

*lll. The quick deterioration of the oleo strut splash guards (para 143)

SPECIFICATION COMPLIANCE

163. The UH-60A helicopter failed to meet the following requirements of the PIDS:

a. Paragraph 3.2.1.1.1.2 - Failed the alternate endurance requirement by 8 minutes (para 22)

b. Paragraph 3.2.1.1.3.1.4. - The vibration levels in several flight regimes exceeded the requirements of this paragraph (para 115)

c. Paragraph 3.7.5.8.1.c - Engine in-flight starts could not be accomplished at the aircraft service ceiling (para 111)

d Paragraph 3.7.6.1 - The forward cyclic control stop is not located as near the cockpit control as possible (para 33)

e. Paragraph 10.3.2.1.1 - The longitudinal cyclic control failed to meet the aft breakout plus friction force by 20 percent (para 32)

f. Paragraph 10.3.2.1.2 - Breakout forces including friction were not symmetrical for the longitudinal and lateral cyclic and collective controls and failed to meet the requirement by 43, 21, and 13 percent, respectively (paras 32, 34, and 36)

*Reported during previous evaluation

g. Paragraph 10.3.2.5 - The directional control would not maintain a zero force position (para 35)

h. Paragraph 10.3.2.5 - The lateral limit control force exceed the requirements by 30 percent (para 34)

i. Paragraph 10.3.3.1.1.1 - The longitudinal, lateral, and directional controls all exhibited stick jump (paras 32, 34, and 35)

j. Paragraph 10.3.7.7.2 - Using full cyclic displacements, aircraft control could not be maintained for left wheel up and nose landings on a 15 degree slope nor on a 12 degree nose down slope (para 78).

164. It is questionable whether or not the UH-60A will meet the requirement of the PIDS, paragraph 3.2.1.1.1.3.(d), in that ground contact may be encountered during a single engine takeoff from a two foot hover.

RECOMMENDATIONS

165. The following recommendations are made:

a. The enhancing characteristics noted in paragraph 160 should be incorporated in future designs

b. The deficiencies reported in paragraph 161 must be corrected

c The shortcomings reported in paragraph 162 should be corrected

d. The deficiences and shortcomings identified during previous evaluations and listed in paragraph 158 should be re-evaluated

e. Consideration should be given to evaluating the following handling qualities on a current production UH-60A (para 30)

(1) Static longitudinal stability in climbs, level and desending flight

(2) Static lateral-directional stability in climbs, level and desending flight

(3) High airspeed maneuvering stability

(4) Short and long term dynamic stability in climbs, level, and desending flight

(5) Controllability at a hover

(6) Single hydraulics failures

f. The following WARNING should be included in chapter 8 of the operator's manual (para 77):

WARNING

During slope operations, vertical clearance between the main rotor tip path plane and the ground is extremely low on the uphill side of the helicopter. Personnel must be warned not to approach or depart the helicopter in the uphill direction.

g. The following WARNING should be included in the instrument flight section of the operator's manual (para 105):

WARNING

The use of radar altimeter settings for level-off commands could compromise flight safety if lower terrain is on the approach end of the runway and the pilot fails to monitor the barometric altimeter. Radar altimeter altitude must not be used to determine arrival at minimum descent altitude or decision height. The level-off mode of the CIS should only be used in the tactical environment where detailed map reconnaisance is conducted prior to IMC flight.

h. The following CAUTION should be included in chapter 8 of the operator's manual (para 81):

CAUTION

Landing from a hover with a relative wind azimuth between 285 and 345 degrees with wind velocity or gusts between 15 and 25 knots should be avoided due to the possibility to encountering a lateral shuffle.

i. The following CAUTION should be included in chapter 8 of the operator's manual (para 155):

CAUTION

Insure both APU compartment access doors are secured prior to operating the main rotor.

j. Consideration should be given to incorporating the following CAUTION in the UH-60A operator's manual (para 60):

CAUTION

The UH-60A has encountered ground resonance when the following three conditions occur simultaneously:

1) parking brake ON

2) aircraft "light" on the wheels

3) extreme aft cyclic control

The ground resonance is characterized by a lateral oscillation (approximately 3 cycles per second). During these oscillations only small attitude changes are encountered; however, roll rate will increase significantly within two seconds. The resonance can be quickly damped by reducing collective, or by bringing the aircraft to a hover, or by centering the longitudinal cyclic control.

k. A brightly colored strip should be painted on the overlapped portion of the right APU compartment access door to reduce the possibility of not securing right door prior to operating the main rotor (para 155)

l. The following NOTE should be placed in the operator's manual (para 27):

NOTE

Rapid collective reductions may result in a transient main rotor speed droop of approximately 2 percent.

m. Paragraph 5-31.2 of the operator's manual should be changed to reflect the maximum slope landing that should be attempted with winds of 3 knots or less is 12 degrees left or right wheel up and 10 degrees nose up or nose down (para 79)

n. Consideration should be given to disabling the heading lockout submode of the CIS (para 98)

o. The automatic course interception capability of the CIS should be expanded to provide intercept information for any selected radial (para 100)

p. The HSI configuration of the CIS should be changed so that the No. 1 and No. 2 needle display could be selectable between doppler, VOR, and ADF (para 99)

q. A positive pilot cueing system to indicate all automatic CIS modes or function changes should be incorporated (para 104)

r. The significance of heat damage during high power hover with the IR subsystem installed should be further evaluated (para 145)

s. A similar engine assessment, as conducted on Project No. 77-58, should be performed on a current production UH-60A (para 83)

t. A redesign of the heading hold subsystem should be considered (para 68)

u. Trim runaways should be evaluated during future tests (para 34)

v. Additional slope landing evaluations should be performed at the extreme cg locations to define the control margins under these conditions (para 76)

w. Limited directional controllability was conducted during this evaluation and consideration should be given to performing additional tests (para 51)

x. Consideration should be given to removing the Red/Remote/White switch on the secondary lights/cockpit floodlight and incorporating the secondary and cockpit floodlight reostats on the battery utility bus (para 137)

y. The pitch bias actuator should be locked in the centered position and phased out of the UH-60A fleet as supplies are diminished (para 113)

z. The low oil pressure indications at low gas producer speeds should be investigated and, if determined to be nondamaging, the cautionary zone on the engine oil pressure gauges and the activation mechanism for the engine oil pressure caution lights should be lowered (para 132)

aa. Future tests should be conducted on a current production UH-60A in order to obtain vibration data that would be more representative of the aircraft (para 115)

bb. Future tests should include maneuvering stability tests at the alternate design gross weight to determine if the vibration levels are as high as that reported during the GCT (para 119)

cc. Consideration should be given to expanding the load factor envelope of the UH-60A and incorporating the envelope in the operator's manual (para 49)

dd. Until the high gross weight load factor envelope is expanded, a "g" meter should be installed in the UH-60A (para 49)

ee. The cockpit installed checklist design should be modified so that it can be changed to reflect current procedures (para 136)

ff. The RAM data on the UH-60A should be reviewed and closely monitored (para 142)

gg. The TGT limiter should be automatically disabled for single engine operation (para 84)

hh. Consideration should be given to reinstalling the forward longitudinal cyclic stick stop to preclude the long term excessive forces having a detrimental effect (para 33).

APPENDIX A. REFERENCES

1. Material Need Document, Action Control No. 10705, February 1972.

2. Letter, AVSCOM, DRSAV-EQ, 29 August 1977, subject: AVRADCOM Test Request 77-17, Airworthiness and Flight Characteristics Evaluation on the YUH-60A.

3. Prime Item Development Specification, Sikorsky Aircraft Division, "DARCOM-CP-2222-S1000D Part 1, "15 October 1979.

4. Technical Manual, TM55-1520-237-10, *Operator's Manual, UH-60A Helicopter. Headquarters Department of the Army*, 21 May 1979, with change 6, dated 7 February 1980.

5. Letter, AVRADCOM, DRDAV-DI, 1 October 1979, Revision 3, 9 January 1980, subject: Airworthiness Release for the UH-60A Black Hawk Helicopter, S/N 77-22716 for Airworthiness and Flight Characteristics Evaluation, Project No. 77-17.

6. Letter, AVRADCOM, DRDAV-DI, 11 April 1980, subject: Airworthiness Release for the UH-60A Black Hawk Helicopter, S/N 77-22717 for Airworthiness and Flight Characteristics Evaluation, Project No. 77-17.

7. Test Plan, USAAEFA Project No. 77-17, *Airworthiness and Flight Characteristics Evaluation, YUH-60A Black Hawk Helicopter*, April 1978.

8. Engineering Design Handbook, Army Material Command, AMC Pamphlet 706-204, *Helicopter Performance Testing*, 1 August 1974.

9. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Stability and Control*, 10 June 1968.

10. Letter, AVRADCOM, DRSAV-D, 6 June 1981, subject: T700-GE-700 Engine Power Available.

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

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

13. Letter, USAAEFA Project No. 77-17-3, 8 Aug 1980, subject: UH-60A Aft CG

14. Military Specification, MIL-F-18372, Flight Control System: Design Installation and Test of, Aircraft: General Specification for, 31 March 1955.

15. Letter, USAAEFA Project No. 78-22, 26 April 1979, subject: Preliminary Airworthiness Evaluation (PAE) III, UH-60A Black Hawk Helicopter.

16. Letter, USAAEFA Project No. 78-11, 4 December 1978, subject: Preliminary Airworthiness Evaluation II A, YUH-60A Helicopter with Production Avionics, Electronic Flight Control System, and Command Instrument Systems.

17. Letter, USAAEFA Project No. 79-11, 20 December 1979, subject: Preliminary Airworthiness Evaluation (PAE) III A, UH-60A Black Hawk Helicopter with the Infrared (IR) Reduction System.

18. DOD Flight Information Publication (Enroute) Flight Information Handbook, 1 July 1981.

19. Letter, USAAEFA Project No. 77-18, 19 January 1978, subject: Preliminary Airworthiness Evaluation, YUH-60A Black Hawk Helicopter with Electronic Flight Control System.

20. Letter, AVRADCOM, DRDAV-DI, 1 November 1979, subject: Preliminary Airworthiness Evaluation (PAE) III, UH-60A Black Hawk Helicopter, USAAEFA Project No. 78-22.

21. Technical Manual, TM 55-1520-237-23-2, *Aircraft General Information Manual, UH-60A Helicopter, Headquarters Department of the Army,* 29 December 1978.

22. Letter, USAAEFA Project No. 77-58, 15 December 1977, subject: T700/ECU Evaluation on YUH-60.

23. Letter, USAAEFA Project No. 78-01, 2 March 1978, subject: UH-60A Command Instrument System Preliminary Evaluation.

APPENDIX B. AIRCRAFT DESCRIPTION

	Paragraph Number
General	1
Airframe	
Fuselage	2-8
Landing Gear	9
Vibration Absorbers	10
Main Rotor	11
Tail Rotor	12
Transmission Systems	13-16
Power Plants	
Engines	17
Engine Power Control System	18
Auxiliary Power Unit	19
Fuel System	20
Electrical System	21-26
Fire Protection System	27-28
Pitot-static System	29
Flight Control System	
General	30
Hydraulic System	31
Automatic Flight Control System	
General	32
Stability Augmentation System	33-35
Flight Path Stabilization System	36-39
Trim System	40-43
Pitch Bias Actuator	44-46
Stabiliator Control System	47-50
Command Instrument System	
General	51
Altitude Hold Mode	52
Heading Hold Mode	53
Go-around Mode	54
Navigation Mode	55
Basic Aircraft Information	56

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, cargo, and weapons during day, night, visual, and instrument conditions. A complete description of the aircraft is contained in the operator's manual (ref 4, app A) and the aircraft general information manual (ref 18, app A). Major features of the helicopter are described below and shown in photos 1 through 3.

AIRFRAME

Fuselage

2. The airframe, as shown in figure 1, includes the nose section, mid-fuselage, aft fuselage, tail cone, tail rotor pylon, stabilator, and main rotor pylon. The fuselage primary structure is aluminum alloy and is semimonocoque in construction.

3. The cockpit nose serves as the avionics compartment and the nose door is constructed of Kevlar. The cockpit roof is one-piece fiberglass incorporating overhead Plexiglas windows. The pilot and copilot windshields are constructed of shatter-resistant glass with electrically operated windshield wipers. The windshields are electrically heated for defogging and anti-icing. The jettisonable cockpit doors are made of Kevlar and contain a sliding window. The pilot's and copilot's seats have a vertical and horizontal adjustment and incorporate a rear tilt feature which allows the seat to be disengaged from its four racks and tilted back into the troop compartment. The seats utilize a five point occupant restraint system. Dual controls and duplicate flight instruments are provided Cockpit drawings are presented in figures 2 and 3. Circuit breaker panels are installed above and behind the pilot's and copilot's heads, just behind the lower console and in the overhead of the cargo compartment. The engine and transmission parameters are monitored on lighted vertical scales, digital readouts, and caution lights.

4. The mid-fuselage section includes the crew chief/gunner stations and troop/cargo compartment. Entrance into this section is through the two piece gunners' windows on each side of the fuselage or through either of the two aft sliding troop cargo doors. The compartment will normally accommodate eleven combat-equipped troops, but can accommodate 14 in a high-density seating arrangement. There are provisions for four litters. Floor load limit is 300 pounds per square foot (lb/ft^2) in the cargo area. The cargo door opening can accommodate a maximum package size of 54 inches high and 68 inches wide.

5. The aft fuselage section connects the mid-fuselage section with the tail cone. This section contains two 181 gallon fuel tanks and two equipment compartments which are accessible from inside the troop/cargo compartment.

6. The tail cone connects the transition section and tail rotor pylon and supports the tail rotor drive shaft and tail pylon. The tail cone also encloses the tail rotor flight controls and tail landing gear. Two Kevlar tail rotor drive shaft covers are hinged to the tail cone top exterior.

7. The tail rotor pylon is supported by and hinged to the tail cone section. It supports the stabilator, intermediate and tail gear boxes and connecting drive shaft,

Photo 1. Front View, UH-60A Helicopter

Photo 2. Left Three-Quarter View, UH-60A Helicopter

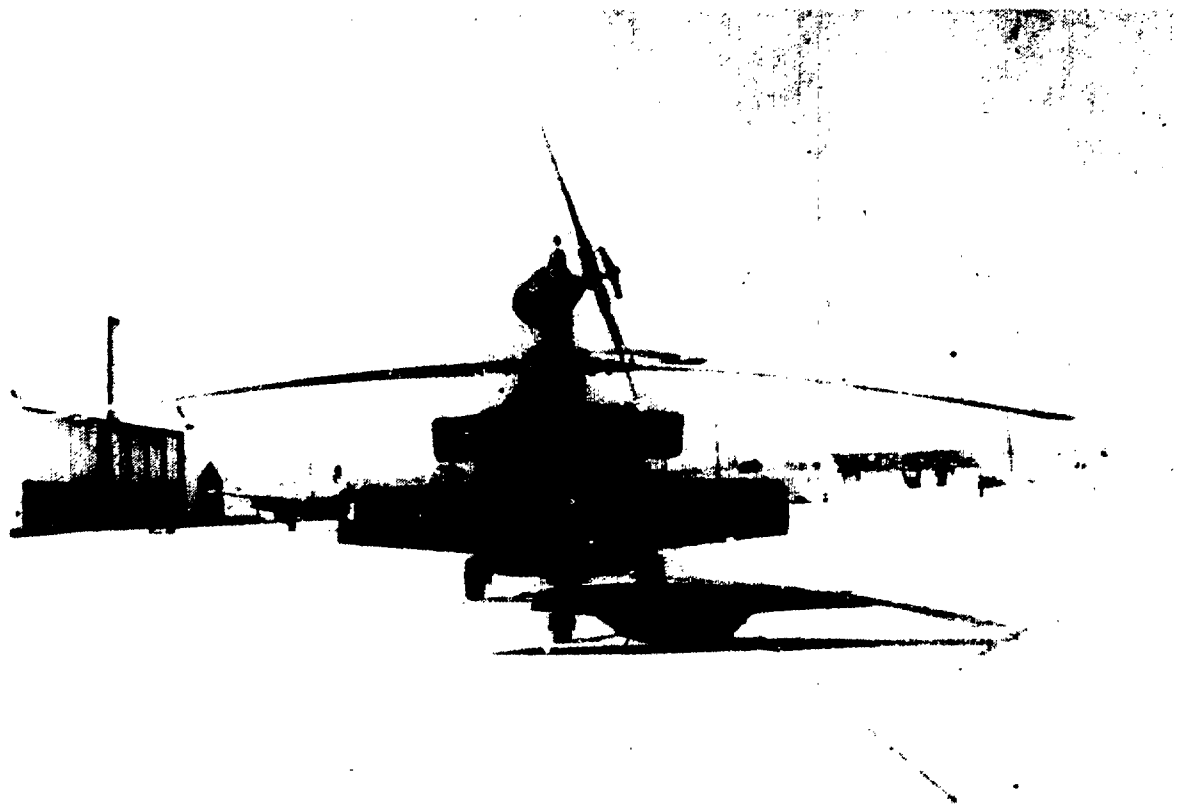


Photo 3. Rear View, UH-60A Helicopter

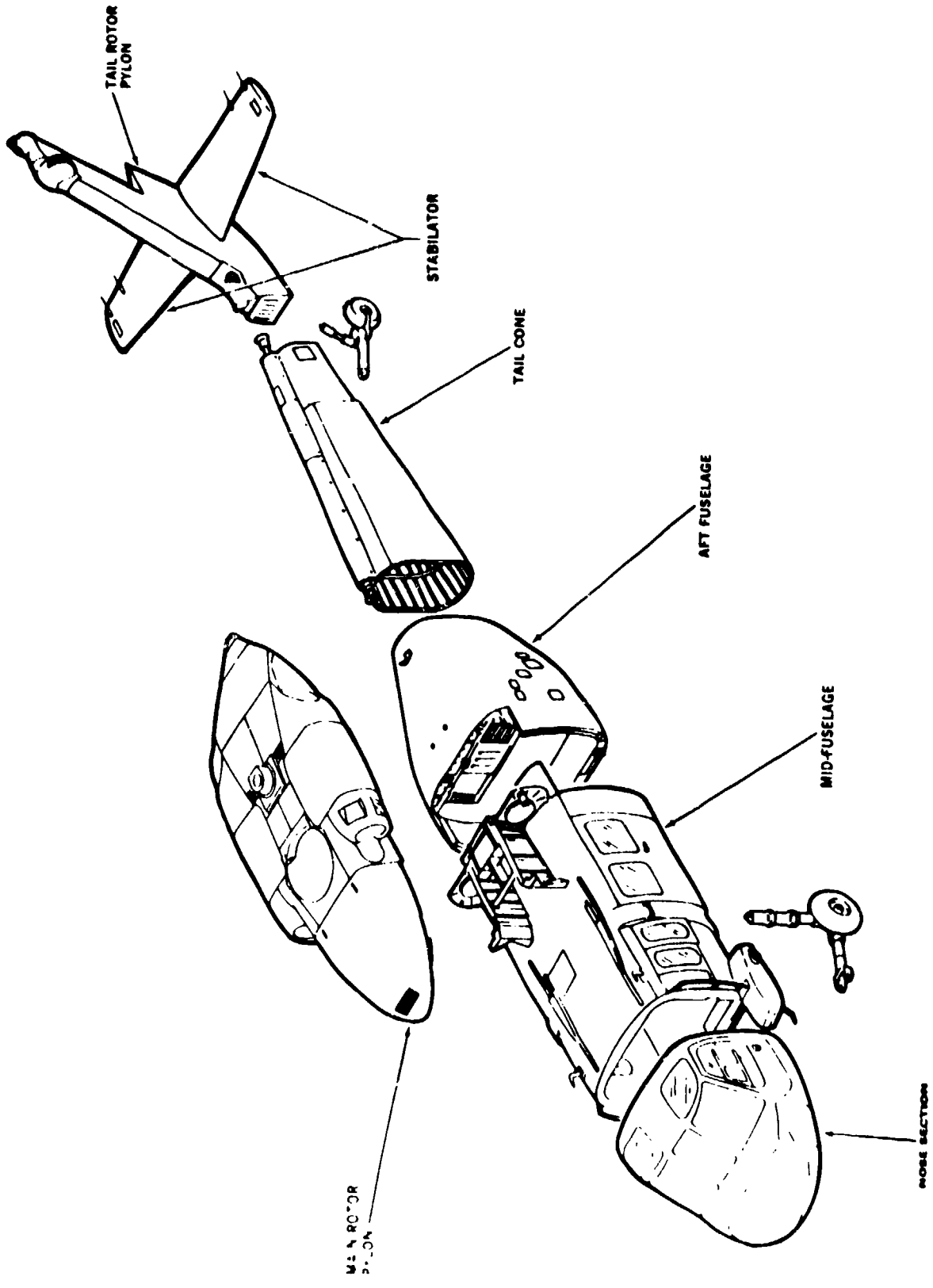
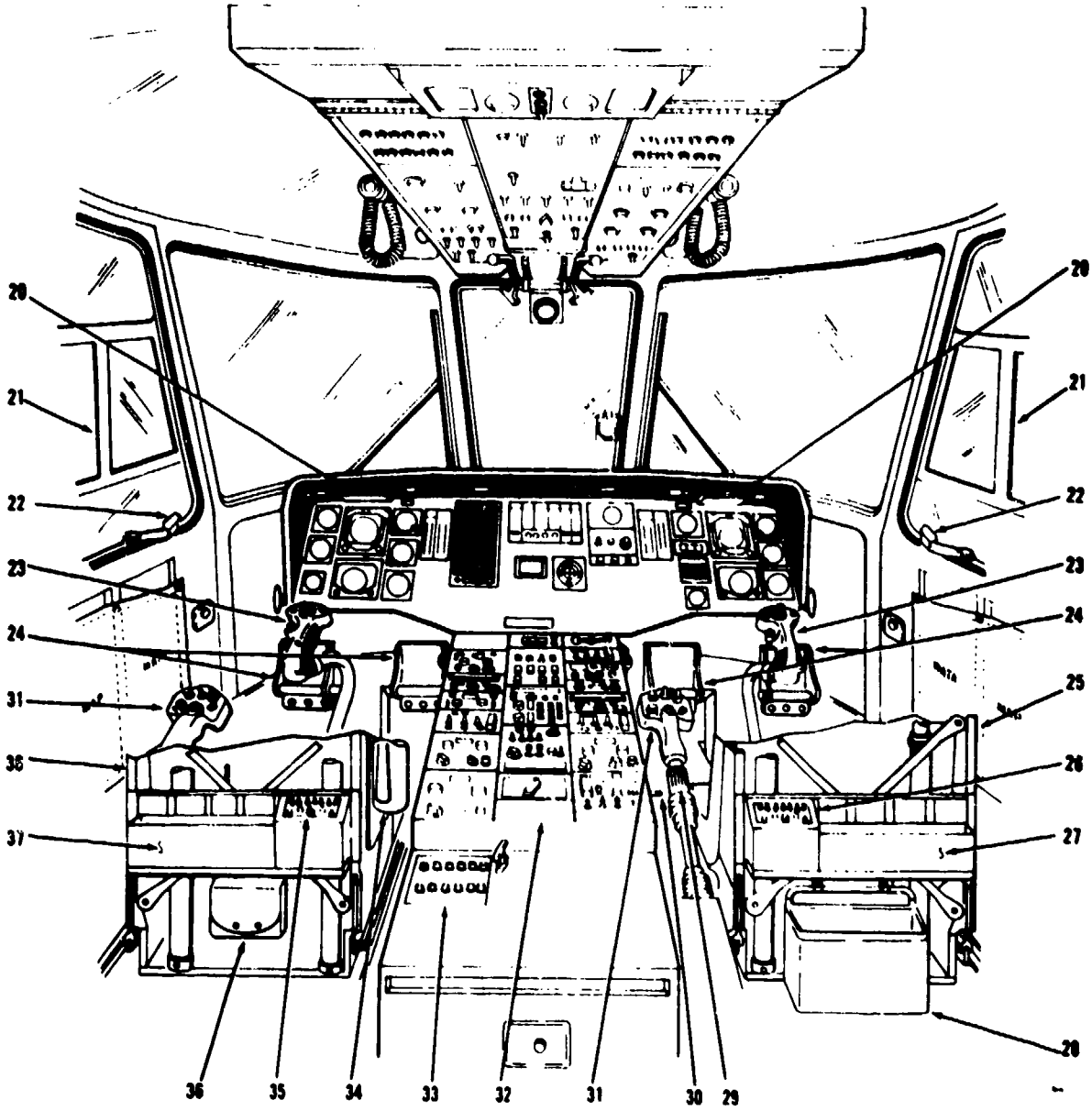
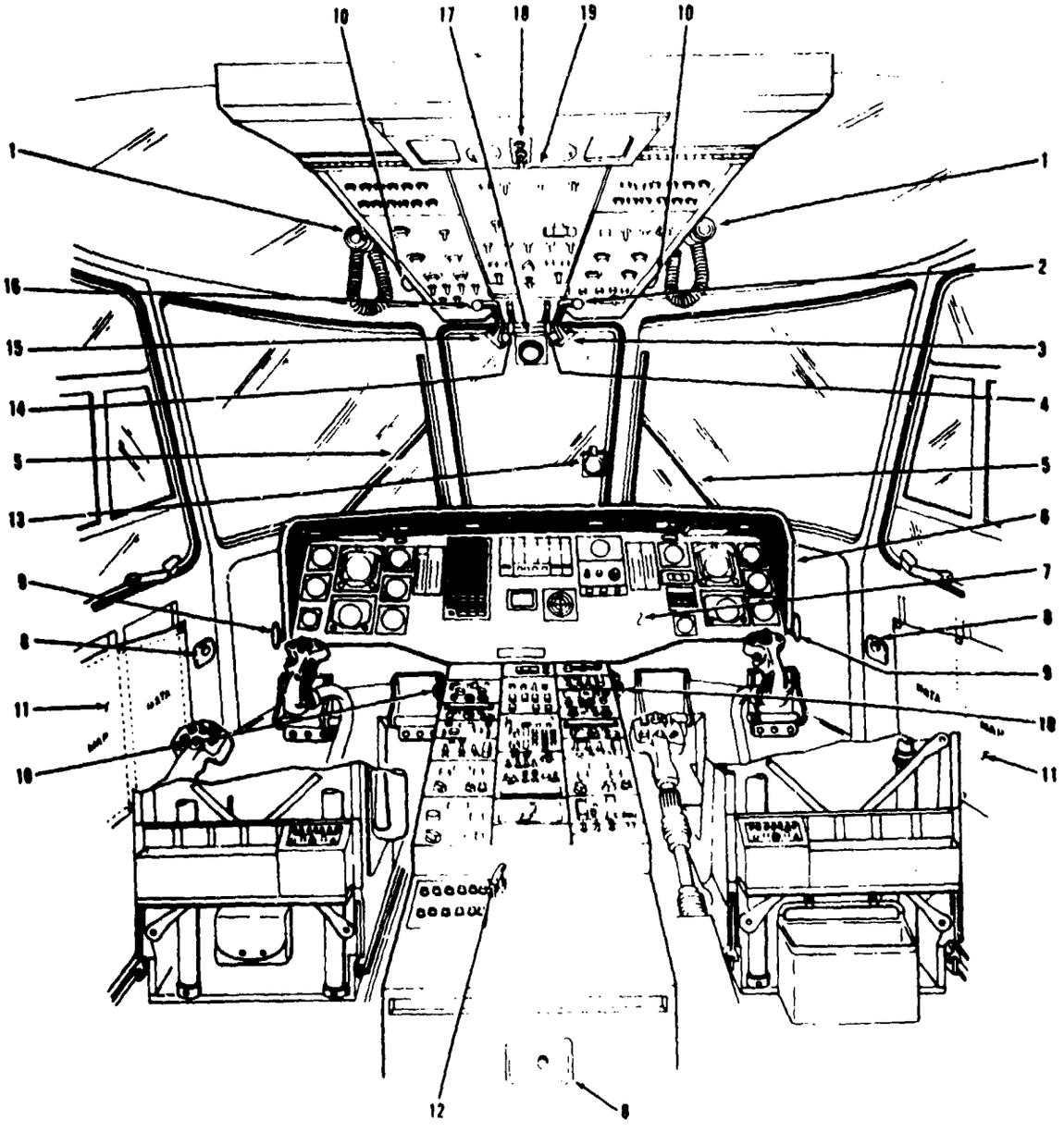


Figure 1. UH-60A Airframe Sections



20 MASTER WARNING PANEL
21 SLIDING WINDOW
22 COCKPIT DOOR EMERGENCY RELEASE
23 CYCLIC STICK
24 DIRECTIONAL CONTROL PEDALS
25 PILOT'S SEAT
26 CREW CHIEF/GUNNER ICS CONTROL PANEL
27 CREW CHIEF AMMUNITION/GRENADE STOWAGE COMPARTMENT
28 STOWAGE BAG
29 COLLECTIVE STICK FRICTION CONTROL
30 ENGINE IGNITION KEY LOCK
31 COLLECTIVE STICK GRIP
32 LOWER CONSOLE
33 BATTERY/BATTERY UTILITY BUS CIRCUIT BREAKER PANEL
34 FIRE EXTINGUISHER
35 GUNNER'S ICS CONTROL PANEL
36 FIRST AID KIT
37 GUNNER'S AMMUNITION/GRENADE STOWAGE COMPARTMENT
38 COPILOT'S SEAT

Figure 2. Miscellaneous Cockpit Furnishings



1 UTILITY LIGHT
2 NO. 2 ENGINE FUEL SELECTOR LEVER
3 NO. 2 ENGINE OFF/FIRE T-HANDLE
4 NO. 2 ENGINE POWER CONTROL LEVER
5 WINDSHIELD WIPER
6 INSTRUMENT PANEL GLARE SHIELD
7 INSTRUMENT PANEL
8 ASHTRAY
9 PEDAL ADJUST LEVER
10 VENT
11 MAP/DATA CASE
12 PARKING BRAKE LEVER
13 STANDBY (MAGNETIC) COMPASS
14 NO 1 ENGINE POWER CONTROL LEVER
15 NO 1 ENGINE OFF/FIRE T-HANDLE
16 NO 1 ENGINE FUEL SELECTOR LEVER
17 FREE-AIR TEMPERATURE GAGE
18 COCKPIT FLOODLIGHT CONTROL
19 UPPER CONSOLE

Figure 3. Miscellaneous Cockpit Furnishings

the tail rotor assembly, and part of the flight controls. With the stabilator removed, the tail pylon can be folded along the right side of the tail cone. The tail pylon leading edge has a built-in frequency modulation (FM) antenna and is hinged to allow inspection of the pylon tail rotor drive shaft. The pylon surface is cambered to unload the tail rotor in cruising flight.

8. The main rotor pylon is constructed of Kevlar and is attached to the upper cabin and aft fuselage. It provides an aerodynamic covering and work platforms for the upper flight controls, main transmission, engines, and auxiliary power unit (APU).

Landing Gear

9. The main landing gear, nonretractable and mounted on each side of the helicopter, incorporates two-stage oleo struts, which will absorb the basic structural design gross weight (16,825 pounds) up to 10 feet per second. The main landing gear wheels incorporate single disc brakes that are toe-operated by the pilots through master cylinders located on the pedals. The nonretractable tail landing gear is secured to a structural attachment fitting in the aft tail cone structure. The tail wheel swivels 360 degrees and can be locked in the trail position.

Vibration Absorbers

10. The UH-60A was equipped with 4 vibration absorbers, one in the nose bay, two in the overhead, and one bifilar located above the rotor system (table 1). All absorbers were tuned for 4/rev vibrations.

Table 1. Vibration Absorber Summary[1]

Absorbers	Part Number	Tuning Weights	Tuning	Fuselage Station
Transmission overhead fwd	70219-02107-044	10.5 pounds	99 pct N_R	317
Transmission overhead (aft)	70219-02107-044	9.5 pounds	99 pct N_R	364
Nose bay	70219-02107-041	0.0 pounds	100 pct N_R[2]	192
Bifilar	70107-08400-042	P N 70107-08404-042[3]	---	341

[1] Butt line for all absorbers was 0.0.
[2] Bucket was out of the recommended tuning range, per Sikorsky recommendations.
[3] Part number for weights on the bifilar.

Main Rotor

11. The articulated four-bladed main rotor consists of a one-piece titanium rotor hub splined to the rotor shaft; a blade retention assembly; elastomeric bearings; blade dampers; adjustable control rods; swashplate assembly; rotating scissors; four titanium-spar main rotor blades composed of a honeycomb core, fiberglass skin, nickel and polyurethane abrasion strips; and a bifilar vibration absorber. The main rotor head incorporates centrifugal droop and anti-flapping assemblies. The main rotor blades are a SC 1095/SC 1095R8 airfoil with slight camber and drooped leading edge, 18 degrees of negative twist, and aft swept tips. An indicator is installed on each blade at the root trailing edge to visually indicate when spar structural integrity (internal pressure) is lost. The blades are attached to the rotor head by two quick-release expandable bolts and all blades can be folded to the rear and downward along the tail cone. The UH-60A is equipped with a gust lock which is designed to prevent the blades from rotating with one engine at idle.

Tail Rotor

12. The four-bladed crossbeam tail rotor has a composite construction and contains no bearings. A pitch change beam on the pitch control shaft provides pitch change motion by deflection of the graphite composite spars. The tail rotor blades are built around two graphite composite spars running tip-to-tip and crossing each other at the center to form the four blades. The two spars are interchangeable and may be replaced individually. Polyurethane and nickel abrasion strips are bonded to the leading edge of the blades. The tail rotor assembly is attached to the tail gearbox on the right side of the pylon and the plane of the rotor is canted 20 degrees up.

Transmission Systems

13. The transmission systems (fig. 4) consists of a main transmission, an intermediate gearbox, and a tail gearbox. Power from the engines is transmitted to the main transmission through input modules.

14. The main transmission is mounted on top of the cabin fuselage just forward of and between the two engines. The main transmission is made of magnesium, has a 3-degree forward tilt, and is modular in construction, consisting of two input modules, the main module, and two accessory modules.

15. The two interchangeable input modules transfer power from the engines to the main module and provide the first gear reduction from 20,900 to 5,750 revolutions per minute (rpm). The input modules are mounted on the left and right front of the main module and support the front of the engines. The input modules are identical and contain an input pinion and gear, and a free-wheel unit. The input bevel pinion gear and quill shaft drives a combining gear which drives the main module. Ramp and roller-type (sprague) free-wheel clutches allow engine disengagement from the transmission during autorotation or in event of a nonoperating engine.

16. An accessory module is mounted on the forward section of each input module. Each accessory module, which is interchangeable, provides mounting and drive for an alternating current (AC) electrical generator and a hydraulic pump package. The rotor speed tachometer sensor is mounted on the right accessory module only. The accessory module will continue to be driven by the main rotor with engine disengagement.

	RPM'S IN	OUT	REDUCTION
MAIN GEAR BOX INPUT MODULE	20903	5748	3.64:1
MAIN GEAR BOX MAIN MODULE	5748	258	22.89:1
INTERMEDIATE GEAR BOX	4116	3319	1.24
TAIL GEAR BOX	3319	1190	2.79
NOTE ALL RPM'S AT 100%			

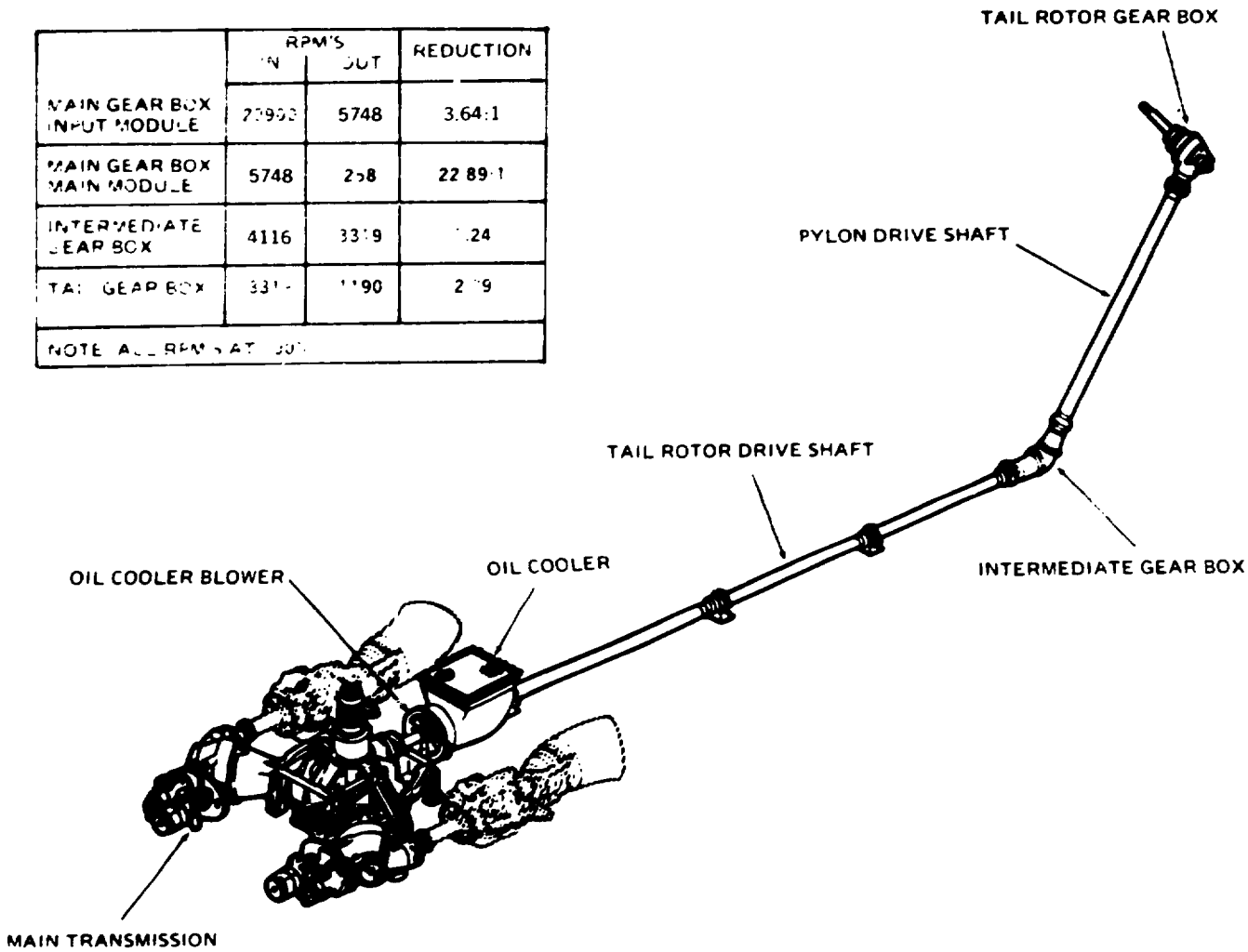


Figure 4. Transmission System Powertrain

POWER PLANTS

Engines

17. The primary power plants for the UH-60A helicopter are two General Electric T700-GE-700 front drive turboshaft engines, rated at 1553 shaft horsepower (shp) at 20,900 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 section, power turbine section, and accessory section (fig. 5). Design features of each engine include an axial-centrifugal 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. The compressor has a bleed-air capability which provides heated air for engine inlet anti-icing and cockpit/cabin heating, and crossbleed engine starting. In order to reduce sand and dust erosion and foreign object damage, an integral particle separator operates when the engine is running. The T700-GE-700 engine also incorporates a history recorder which records total engine events. 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
Variable geometry	Inlet guide vanes, and stages 1 and 2 stator vanes
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 lb max
Length	47 in.
Maximum diameter	25 in.
Fuel	MIL-T-5624 grade JP-4 or JP-5
Lubricating Oil	MIL-L-7808 or MIL-L-23699

Engine Power Control System

18. The engine control system consists of a load demand system, power control system, and a speed control system. The load demand system supplies a collective signal to the load-demand spindle on the engine hydromechanical unit (HMU) which controls fuel flow. The power control system for each engine controls the HMU through the engine power control levers in the cockpit. The engine speed control system operates through the ENG RPM switch on the collective and the electrical control unit (ECU). The ECU receives inputs from the engine alternator, thermocouple harness, power turbine (N_p) sensor, torque and overspeed sensor, torque signal from the opposite engine for load matching, and feedback signals from the HMU for system stabilization, and a demand speed from the ENG RPM switch. The temperature limiting system limits fuel flow when the requirement is so great the turbine gas temperature (TGT) reaches 840°C ±5°. Fuel flow is reduced to hold a constant TGT. The overspeed system provides both gas generator (N_g) and N_p overspeed protection. Fuel flow is reduced if N_g reaches 102 percent or N_p reaches

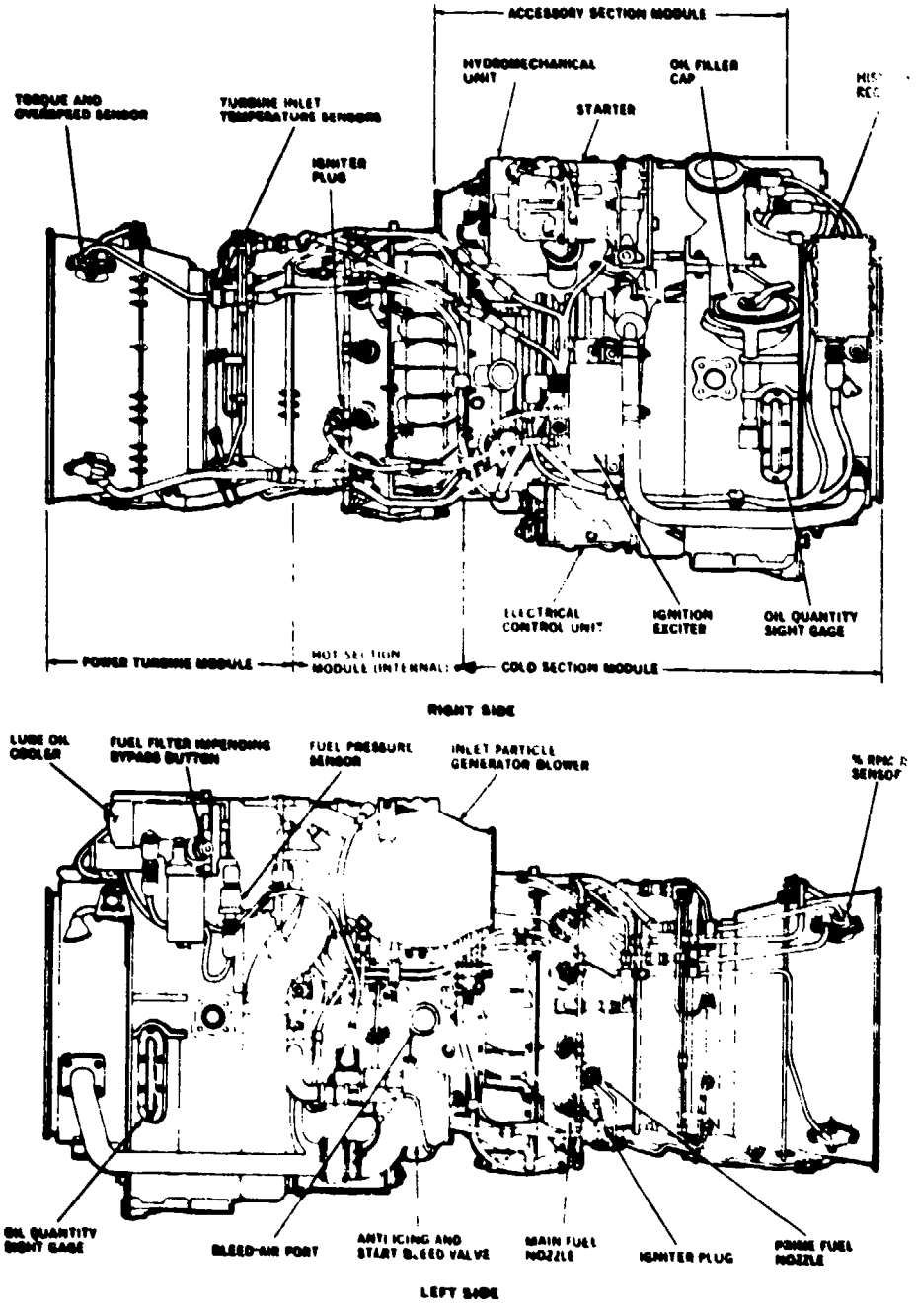


Figure 5. F100-GE-100 Engine

106 percent. If the ECU fails it can be overridden by placing the engine power control level in the LOCKOUT position and using it as a throttle. The ECU can be reset by moving the power control lever to IDLE.

Auxiliary Power Unit

19. The SOLAR module T-62T-40-1 APU consists of a gas turbine power section, a reduction gear drive, and appropriate controls and accessories. The gas turbine power section uses a single centrifugal compressor and a single-stage radial inflow turbine mounted back-to-back on the end of the high speed shaft. Power extracted from the turbine drives the compressor and high speed output shaft. The combustor is an annular type with six air-atomizing fuel injection points. Ignition is accomplished with a separate pressure atomizing fuel nozzle and a spark plug. A pinion gear supported by three planetary reduction gears (5.13:1) reduces turbine shaft speed from 61,565 rpm to an output speed of 12,000 rpm. The APU system provides pneumatic power for main engine starting and cabin heating, electrical and hydraulic power for ground operations, and inflight emergency electrical power. APU system accessories include a prime/poost pump, hydraulic accumulator, hydraulic handpump, hydraulic utility module, hydraulic backup pump, alternating current generator, and hydraulic start motor. Pertinent information concerning the APU is listed below.

Rated engine speed	61,565 rpm
Exhaust gas temperature	1200° ± 10°F
Weight (dry)	92 lb (approx)
Output shaft horsepower (shp)	90 shp
Fuel consumption at rated power	150 lb/hr (approx)
Reduction gear and accessories	
Input speed (rated)	61,565 rpm
Output speed (rated)	12,000 rpm
Fuel control assembly	4235 rpm
Fuel	JP-4, JP-5
Lubricating oil	MIL-L-7808 or MIL-L-23699
Oil pump capacity	3 quarts
Components:	
Compressor	Single-stage, centrifugal flow
Turbine	Single-stage, radial flow
Combustor	Annular type

FUEL SYSTEM

20. A separate suction fuel system is provided for each engine and consists of two interchangeable, crash-worthy, ballistic-resistant tanks with a total capacity of 362 gallons, self sealing lines, firewall mounted breakaway selector valves, engine-driven suction pumps, and a prime/boost pump. The prime pump primes all fuel lines if prime is lost and acts as an APU boost pump for APU operation. The fuel tank selector valves are manually operated through two engine fuel system selector levers on the cockpit overhead engine control quadrant. The lever positions are OFF (closing the valve), DIR (direct - allowing fuel used by each engine to be taken

from its respective tank), and XFD (crossfeed - allowing the left engine to use fuel from the right tank or the right engine to use fuel from the left tank). Fuel quantity in each tank is continuously displayed on a vertically lighted indicator with a digital total fuel quantity readout. Each tank contains a low-level sensor which activates two low-level warning lights on the caution-advisory panel indicating #1 FUEL LOW or #2 FUEL LOW when the fuel level decreases to 170-190 pounds each at a cruise attitude. A range extension kit may be added to the aircraft which provides approximately 780 gallons of additional fuel.

ELECTRICAL SYSTEM

21. The UH-60A uses AC as primary electrical power which is supplied by two independent primary AC power systems (No. 1 and No. 2 main generator), or an auxiliary AC power backup system (APU generator). The UH-60A also has provision for an external AC power supply. An electric power priority feature allows either the No. 1 or No. 2 main generator, which in turn automatically preempts external power. Primary direct current (DC) power is obtained from two converters, with the battery as a secondary DC power source.

22. The primary AC power system delivers regulated three phase, 115/200 volt AC, 400 hertz (Hz) to the aircraft. Each system consists of an oil-spray cooled, brushless, 30/45 kilovolt-ampere (kVA) generator mounted on and driven by the transmission accessory gear box module, a current transformer, a generator control unit, and a current limiter, all of which are interchangeable. The No. 1 and No. 2 generator system outputs are applied to the No. 1 and No. 2 AC primary buses respectively with either system capable of carrying the entire load in the event of a failure. Each generator's output is monitored by the generator control unit and current transformer which will disable the generator due to overvoltage, undervoltage, feeder fault, underfrequency, and transient current overload conditions. The underfrequency protection is disabled with the helicopter weight off the wheels through a microswitch on the left landing gear.

23. The auxiliary AC power system provides power for ground checks or may be used in the event both primary AC systems fail. The auxiliary power system consists of a 115 volt AC, three-phase, 40 Hz, 20/30 kVA air-cooled generator mounted on and driven by the APU, a current transformer, and a generator control unit. If the APU generator is the sole source of AC generated power, all equipment may be operated except when the backup pump is on the windshield anti-ice is prevented from being used.

24. An external AC power connector is located on the right side of the helicopter and accepts a ground source of 115 volt, three-phase AC power. External power is supplied to the aircraft systems by placing the EXT PWR switch, located on the overhead console, to RESET then ON. If an overvoltage, undervoltage, overfrequency, or underfrequency fault occurs, external AC power will be disconnected from the electrical buses.

25. Two 200-ampere converters, each normally powered by the No. 1 and No. 2 AC primary buses, convert AC into DC power and reduce it to 28 volts. If one converter's output is lost, the converter load will be transferred to the operating system and the appropriate converter caution light will illuminate.

26. A 24-volt DC, 5.5 amperehour, 20 cell nickel-cadmium battery provides secondary or emergency DC power. It supplies DC power to the battery and battery utility buses for operating essential DC powered equipment in the event of a dual converter failure and its operation is controlled through the battery switch on the overhead console. The battery will supply power to the DC essential bus if both converters have failed and the battery is at least at 35 percent charge. A charger-analyzer restores battery charge and determines the condition of the battery. A BATTERY LOW CHARGE caution light will illuminate when the charge lowers to 40 ±5 percent of battery capacity. At 35 ±5 percent of battery capacity the DC essential bus will be disconnected from the battery. If the internal temperature of the battery reaches 70°C or if a battery cell dissimilarity condition exists, a BATTERY FAULT caution light will illuminate and the battery will automatically be disconnected from the charging circuit.

FIRE PROTECTION SYSTEM

27. The engine and APU compartments are monitored by infrared radiation (IR) sensing units and protected by a high-rate discharge extinguishing system. The detection system consists of two IR detectors in each engine compartment and one in the APU compartment, three control amplifiers, and a test panel. If a fire is sensed in any of the three compartments the sensors will illuminate the appropriate "T" handle in the cockpit as well as the fire warning light on the master caution panel. Actuation of a "T" handle by the crew arms the fire extinguishing system and shuts off fuel. The extinguishing agent is discharged into the compartment which has the activated "T" handle by positioning the fire extinguisher toggle switch, located on the overhead control panel, to MAIN or RESERVE. A fire detector test on the overhead control is used to verify proper operation of system components.

28. Two containers each filled with 2.5 pounds of liquid bromotrifluoromethane and charged with gaseous nitrogen are mounted behind and to the right of the APU compartment. Both containers have dual outlets, each with its own firing mechanism. A crash-actuated system is incorporated into the fire extinguisher system which, upon an impact of 10 g or more in any direction, automatically fires both extinguishers into both engine compartments.

PITOT-STATIC SYSTEM

29. The UH-60A has a dual pitot-static system. The two electrically heated pitot tubes (with static ports) are aft and above the pilot's and copilot's doors. The right (No. 2) pitot tube provides ram pressure to the pilot's airspeed indicator and the left (No. 1) pitot tube provides ram pressure to the copilot's airspeed indicator. The static sources for the two systems are interconnected and provide static pressure to both pilot's airspeed indicators, altimeters and, vertical velocity indicators. In addition to standard cockpit instrumentation, ram and static pressures are converted into electronic airspeed signals by an airspeed transducer and an air data transducer to be utilized by the Automatic Flight Control System (AFCS) and Command Instrument System (CIS).

FLIGHT CONTROL SYSTEM

General

30. The UH-60A utilizes conventional helicopter cyclic, collective, and directional controls powered by a triply redundant 3050 PSI hydraulic system. The pilot and copilot controls are dual but have separate paths to a combining linkage for each control axis. The control inputs from the cockpit controls are carried by mechanical linkage through the pilot-assist servos/actuators to a mixing unit. The mixing unit combines, sums, and couples the cyclic, collective, and yaw inputs (fig. 6) and provides proportional output signals to the main and tail rotor controls (fig. 7). Pilot control is assisted by an AFCS comprised of five basic subsystems: Stabilator, Pitch Bias Actuator (PBA), Stability Augmentation System (SAS), Trim System, and Flight Path Stabilization (FPS).

Hydraulic System

31. The UH-60A has two separate hydraulic systems, a first stage and second stage, and incorporates a third hydraulic pump/reservoir capable of pressuring either the first or second stage systems if required. The components of the hydraulic systems are three hydraulic pump modules, two transfer modules, a utility module, three dual primary servos, one dual tail rotor servo, six pilot-assist servos, an APU accumulator, an APU handpump, and a servicing handpump. There are three hydraulic pressure supply systems, number 1, number 2, and backup hydraulic pump modules. All are completely independent and each is fully capable of providing essential flight control pressure. The first and second stage pump modules are driven by the main gearbox and supply pressure to the flight control servos whenever the main rotor is turning. The number 1 pump module supplies 3050 pounds per square inch (psi) to the first stage of the three main rotor (primary dual) servos and to the first stage of the tail rotor servo. The number 2 pump module supplies 3050 psi to the second stage of the three main rotor (primary dual) servos, 3050 psi to the collective and yaw boost servos, the SAS actuators, and pitch trim actuators. The electrically operated backup hydraulic pump supplies emergency hydraulic pressure to the first and/or second stage hydraulic systems. This system can also supply hydraulic pressure to all servos during ground checkout and recharges the APU accumulator. The electric motor driving the backup pump module is automatically activated by either a low-pressure sensing switch in the number 1 and number 2 pump modules, by the APU start accumulator switch, or by the manual switch in the cockpit. A leak detection/isolation feature is built into the hydraulic system, using fluid quantity switches on the pump modules, check valves and shutoff valves in the transfer modules, and electronic logic modules. When a fluid quantity switch senses a fluid loss in a system, the logic module will shut off the required valve or valves to isolate the leak and turn on the backup pump. A simplified hydraulic system schematic is presented in figure 8.

Automatic Flight Control System

General:

32. The Sikorsky UH-60A AFCS (fig. 9) is designed to enhance helicopter stability and handling qualities. The system consists of five major subsystems: The SAS, FPS system, trim system, PBA, and stabilator control system. Electronic control of the systems is provided by commands from a digital SAS/FPS computer and a SAS

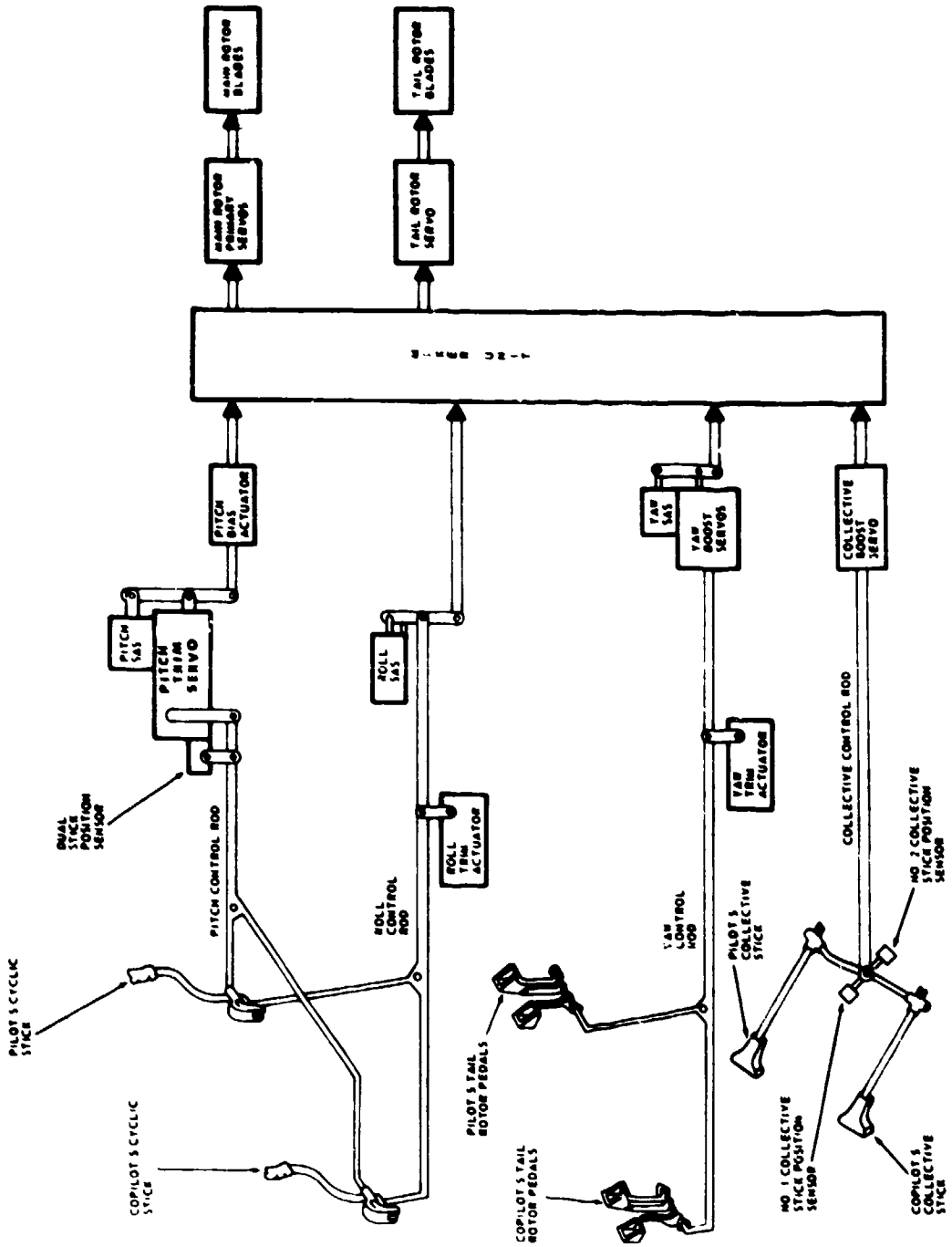


Figure 6. Flight Controls Linkage, Simplified Diagram

MECHANICAL

FROM	TO	REASON
COLLECTIVE	YAW	ANTI-TORQUE
COLLECTIVE	LATERAL	LATERAL LEAD (RIGHT TRANSLATION)
COLLECTIVE	LONGITUDINAL	ROTOR DOWNWASH ON STABILATOR
YAW	LONGITUDINAL	TAIL ROTOR LIFT VECTOR

ELECTRICAL

COLLECTIVE	YAW	CAMBER OF VERTICAL STABILIZER VARIES SIDE LOAD WITH AIRSPEED.

Figure 7. Flight Control Compensations Chart

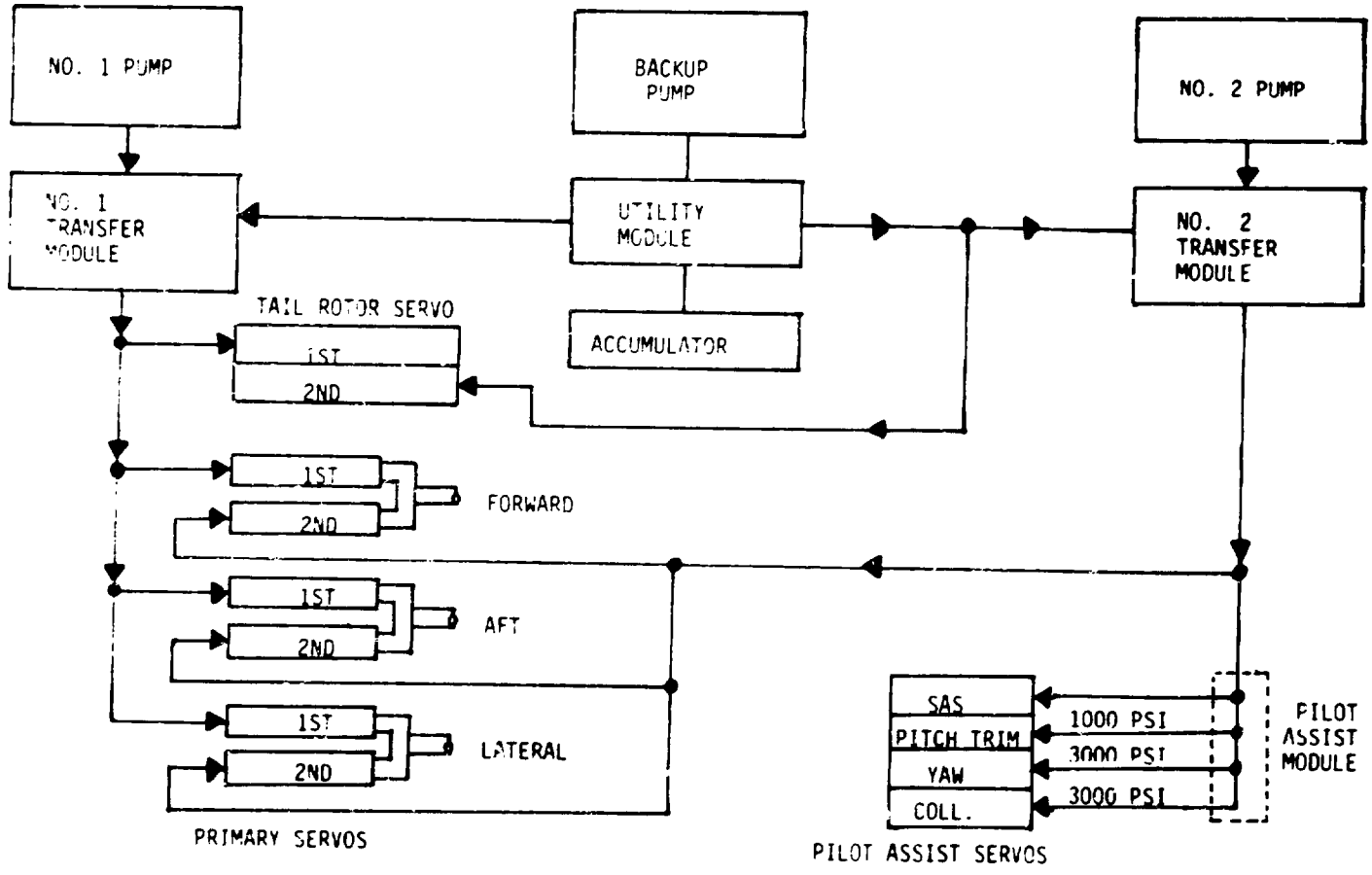


Figure 8. Hydraulic System Block Diagram

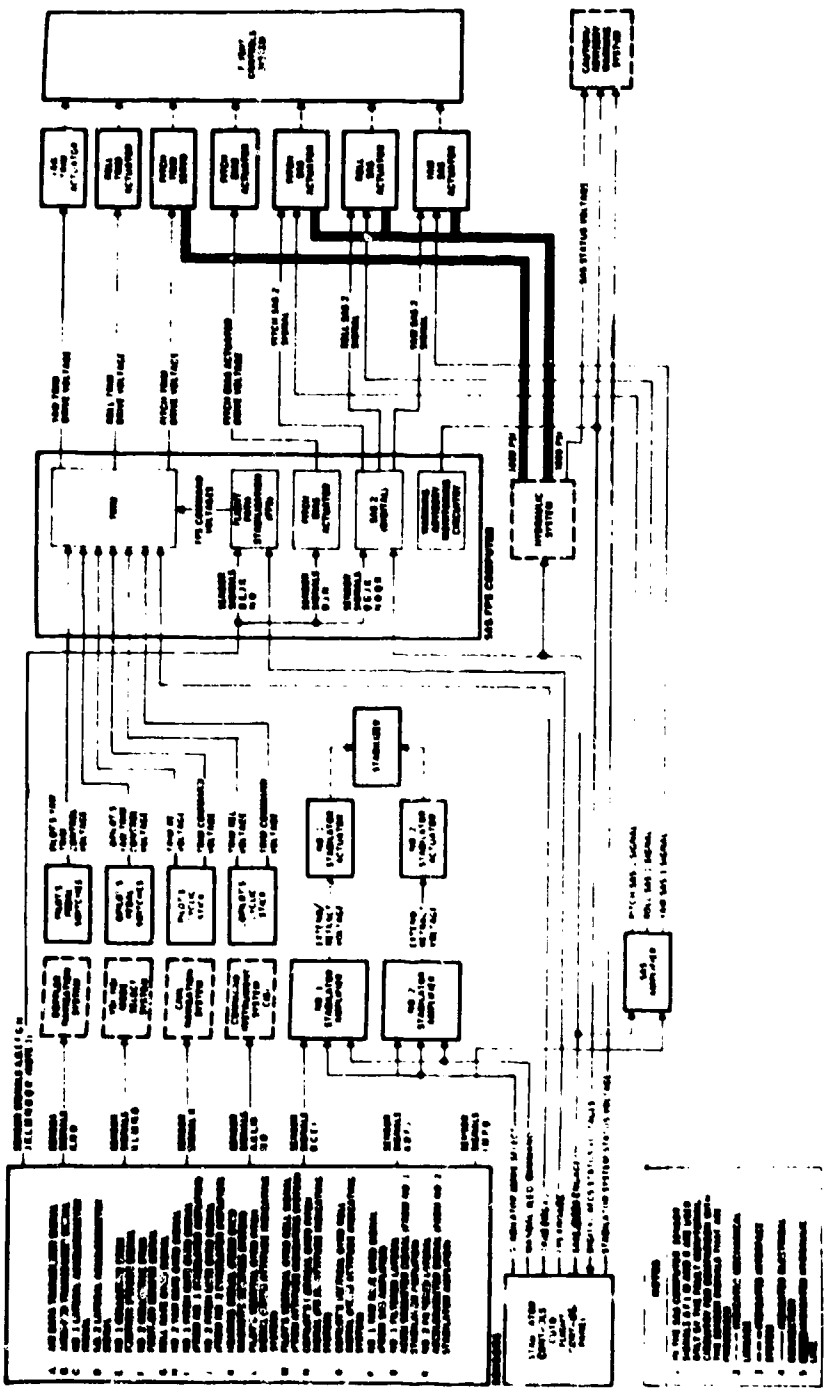


Figure 9. AFCS Functional Block Diagram

analog amplifier. The SAS provides three-axis rate damping, pseudo attitude retention, and limited turn coordination. The FPS provides three-axis attitude and airspeed hold and is the primary source of automatic turn coordination. The trim system provides control position hold and control forces versus position gradients. The PBA is designed to provide positive static longitudinal stability and contributes to positive maneuvering stability. The stabilator control system automatically positions the stabilator as a function of flight parameters to tailor aircraft pitch attitude and dynamic response.

Stability Augmentation System:

33. The SAS functions to provide three-axis rate damping and pseudo attitude retention. The SAS is a dual system with one subsystem (SAS-1) controlled by the analog SAS amplifier and one subsystem (SAS-2) controlled by the digital SAS/FPS computer. It is redundant in sensors and command signal path; however, both SAS subsystem command signals drive a single SAS actuator in each axis. During normal operation with both SAS-1 and SAS-2 engaged, each provides one-half of system nominal gain and one-half of total system control authority. The control authority of each is electrically limited to ±5 percent of total control travel in pitch, roll, and yaw. SAS inputs to the SAS servo valves are additive to provide a total authority of 10 percent. The sum is limited to ±10 percent authority by mechanical limits of SAS actuator travel. Selectable operation of either SAS-1 or SAS-2 is available at the center console and switching either subsystem OFF automatically doubles the gain of the remaining SAS while its authority remains at 5 percent. All three axes provide rate damping and lagged rate damping (pseudo attitude retention). A washout of the rate damping signal is incorporated in the pitch and yaw channels to prevent saturation during a steady turn.

34. The SAS-1 is controlled by the SAS-1 analog amplifier which continuously derives commands based on inputs from the No. 1 yaw rate gyro, the No. 1 pitch rate gyro, and a roll rate signal derived from the No. 2 vertical gyro, and the No. 1 filtered lateral accelerometer signal. The SAS-2 is controlled by the SAS/FPS digital computer. SAS-2 commands are continuously generated in response to signals from; the roll rate gyro, No. 2 pitch rate gryo, and signals derived from magnetic compass gyros (yaw rate), No. 1 vertical gyro (pitch and roll rate), and, No. 1 filtered lateral accelerometers. At airspeeds above 60 knots indicated airspeed (KIAS), input signals from the No. 1 filtered lateral accelerometer and the No. 1 vertical gyro (derived rate) are provided to the SAS-2 system to stabilize yaw during coordinated turns.

35. SAS-2 operation is continuously monitored by the SAS/FPS computer. This monitor system compares inputs from independent sources and compares SAS command to SAS actuator output. Failure of any of these comparison checks in SAS-2 input or output indicates a SAS-2 failure (pitch, roll, or yaw channel) and the control input from the affected channel will be removed (actuator remains at failed position) and the SAS-2 advisory light will be illuminated. SAS-1 does not contain fault detection logic.

Flight Path Stabilization System:

36. The FPS is primarily an aircraft attitude hold system that incorporates conditional capability for airspeed hold and turn coordination. The FPS works through the roll, pitch, and yaw trim actuators. The FPS can drive the cockpit

control to any position to which the pilot/copilot can trim the controls, resulting in a 100-percent FPS parallel control authority. The AFCS limits the rate of FPS within the maximum override force limits stated in the trim system section. Since FPS inputs drive the cockpit controls through the trim actuators, the TRIM must be ON in order to have FPS.

37. The attitude hold function of the FPS is designed to maintain a desired heading or pitch and roll attitude. The trim attitude, once established, is automatically maintained unless changed by the pilot. At airspeeds greater than 60 KIAS the pitch axis of the FPS seeks to maintain the airspeed for which the trim attitude has been established. When the reference pitch attitude is changed a time delay in the airspeed hold function allows time to stabilize at the new trim airspeed prior to initiating the airspeed hold function. During this time the attitude hold function maintains the pilot-selected pitch atttiude.

38. The FPS provides two yaw channel functions: heading hold and automatic turn coordination. For heading hold (below 60 KIAS), the aircraft is maneuvered to the desired heading with the pilot's or copilot's feet depressing one or both of the pedal switches. When the pilot or copilot removes his feet from the switches the aircraft automatically maintains that reference heading. At airspeeds greater than 60 KIAS the coordinated turn feature of the FPS is operational. The coordinated turn feature is initiated by a lateral stick displacement of approximately 1/2 inch and a bank angle of greater than 2 degrees. The feature is disengaged when the bank angle is less than 1 degree and the roll rate has decreased to below 2 degrees per second. Turn coordination is accomplished by directional control inputs through the trim actuator to zero the side force as sensed by the lateral accelerometers in the stabilator control system. At airspeed greater than 60 KIAS, heading hold is auto matically engaged unless the pilot engages the turn coordination feature.

39. The FPS and all inputs are subject to a number of cross-checks, within the computer. In essence, each input (*i.e.*, attitude, rate, airspeed, etc.) is compared either against another independent source of the same information or, in the case of rate inputs, a computer-derived rate. If these comparisons exceed the preprogrammed tolerance, the particular FPS function will be disabled and the appropriate AFCS advisory light and the FPS FAIL caution light will be illuminated.

Trim System:

40. The trim system provides zero force control centering at a pilot/copilot selected trim control position, a spring breakout force plus gradient and a pedal damper force. The trim system is selected by activating the push-on push-off switch, marked TRIM, on the AFCS control panel.

41. With the trim system selected OFF there is no control force gradient or control centering in the cyclic control system or directional control system. Directional control movements will be resisted by a pedal damper which generates an opposing pedal force opposite to the proportional rate of pedal movement. This damping force is electrically generated but is continuously engaged without regard to TRIM switch position. With the trim system ON, directional and lateral control forces are developed in the electromechanical trim actuators. These actuators incorporate an electrically controlled rotary spring assembly which allows the pilot to select the zero force control trim position. The designed maximum override force full opposite

control position, is 80 pounds in directional and 19 pounds in lateral cyclic control. Longitudinal cyclic control forces are developed in an electrohydromechanical pitch trim actuator with a designed maximum override force of 20 pounds.

42. With the trim system selected ON and FPS OFF, the pilot/copilot may change the cyclic control trim position through two means: a cyclic trim release switch and cyclic beep trim switch. The cyclic beep trim switch allows the cyclic control trim position to be changed in one direction at a time, at a fixed rate of travel, by electrically driving the trim actuator through the rotary spring assembly. The beep trim switch is a four-position "chinese hat" switch mounted on the cyclic stick grip. Activation of the trim release button switch released the force gradient on the longitudinal and lateral cyclic. The position of the cyclic control when the trim release switch is open (released) becomes the new cyclic trim position. At airspeeds below 60 KIAS, when the pedal switches are closed (any pedal switch depressed), the electronically controlled yaw force gradient spring is repositioned by pedal movement resisted only by the pedal rate damper. When the pilot/copilot removes his feet from the pedals which release the pedal switches, the electronically controlled rotary spring reengages, holding the pedals at the new trim position through the pedal breakout plus gradient spring. Above 60 KIAS the pedal switches and the TRIM REL switch together provide yaw trim release.

43. The SAS/FPS computer monitors the trim system by comparing the commanded trim actuator position to the actual position in all three axes. (Trim actuator position may be commanded by the pilot or by the FPS.) If this comparison is out of tolerance, the trim system is shut off in the defective axis and the TRIM FAIL caution light and TRIM advisory light on the AFCS computer are illuminated. The trim system may be reset by pressing both POWER ON RESET buttons on the AFCS control panel.

Pitch Bias Actuator:

44. The PBA is an electromechanical differential actuator built into the longitudinal cyclic control system to assure a stable gradient of longitudinal cyclic versus airspeed. It receives airspeed, pitch attitude, and pitch rate inputs from the SAS/FPS computer continuously whenever power is applied to the aircraft and the SAS/FPS computer detects no faults prejudicial to PBA function regardless of AFCS control panel switch condition. Airspeed signals do not effect the PBA operation below 80 KIAS. PBA inputs do not feed back to the cockpit controls. The PBA is, in effect, a variable length control rod which changes the relationship between longitudinal cyclic control and swashplate tilt as a function of flight parameters.

45. The authority of the PBA is 15 percent of longitudinal cyclic full throw and is limited by the computer to a maximum rate of 3 percent per second. PBA function is monitored by the SAS/FPS computer by an actuator feedback system. If actuator position differs from the commanded position by more than the predetermined tolerance, power is removed from the PBA, the actuator remains in the position it was in at the time of failure, and the PITCH BIAS FAIL caution light is illuminated. This could result in loss of up to 15 percent (1.5 inches) of forward or aft cyclic control authority. Intermittent PBA failures due to an actuator position versus command "no compare" may be reset by pushing both POWER ON RESET buttons on the AFCS control panel.

46. The PBA operation may be failed or degraded by "no compare" results in airspeed, pitch rate, vertical gyro inputs, internal mechanical failure, or various SAS/FPS computer failures. A pitch rate or vertical gyro failure results in the PBA centering. An airspeed failure results in a constant 120-knot airspeed signal. A mechanical failure of the PBA causes the actuator to remain in the position in which it failed.

Stabilator Control System:

47. The stabilator control system is an electrically controlled and activated system. The primary purposes of the system are to schedule stabilator incidence to eliminate excessively nose-high attitudes at low airspeed due to downwash impingement on the stabilator, and to optimize pitch attitudes for climb, cruise, and autorotational descent. The control system is composed of two analog amplifiers which operate from independent input sources and command the position of two electric jackscrew actuators acting in series. During normal operation these jackscrews operate in unison, with each providing one-half of the stabilator position input.

48. The stabilator position is programmed between ±2 degrees trailing edge up and 38 ±4 degrees trailing edge down as a function of four variables: airspeed, collective, control position, pitch rate, and lateral acceleration. The airspeed input primarily allows the stabilator to align with the main rotor downwash during low-speed flight, thus reducing the stabilator download and eliminating excessively nose-high pitch attitudes at low airspeed. The collective control input reduces coupling of pitch attitude to collective in forward flight. Pitch rate and lateral acceleration inputs are designed to improve the dynamic response of the airframe. Pitch rate inputs to the stabilator system provide a degree of pitch rate damping to supplement SAS-commanded damping. The lateral accelerometer inputs, providing an indication of both side force and yaw angular acceleration, decouple the pitch response to tail rotor thrust changes resulting from changes in the inflow through the tilted tail rotor with sideslip variation.

49. The stabilator system is independent of the other AFCS subsystems although it shares common inputs, collective position, airspeed, and lateral acceleration inputs are all dual inputs which are compared in the AFCS computer. The output of the No. 2 pitch rate gyro is compared with a pitch rate derived in the AFCS computer. If the AFCS computer detects a "no compare" in those inputs, the appropriate caution/advisory lights will be illuminated and affected AFCS computer controlled functions will be shut down; however, the AFCS computer effects no control over the stabilator system fction.

50. Stabilator malfunctions are detected and controlled within the stabilator amplifier system. The positions of the two actuators are monitored and compared by rate and position. Any system malfunction which causes a minimum difference in actuator position (10 degrees at airspeeds less than 30 KIAS and 4 degrees a airspeeds greater than 150 KIAS) results in an automatic shutdown of power to both actuators. If the malfunction is transient, the stabilator system may be reset by pressing the stabilator AUTO CONTROL RESET button on the AFCS control panel. The pilot may at any time take manual control of the stabilator and control its position by referring to cockpit-mounted stabilator position indicators.

COMMAND INSTRUMENT SYSTEM

General

51. The Command Instrument System (CIS) (fig. 10) is an integrated navigational system based on a Command Instrument System Processor (CISP) that processes signals from navigation and flight instruments to display pitch, roll, and collective position commands on the pilot and copilot Vertical Situation Indicators (VSI). The major components of the CIS are the CISP and the Command Instrument System/Mode Selector panel (CIS/MS). Related equipment includes; two VSI's, two Horizontal Situation Indicators (HSI), the go-around (GA) switches (one on each cyclic stick) and the pilot and copilot VSI/HSI mode select panels. The pilot and copilot VSI/HSI Mode Selector panel is used to select the type of navigational or approach data to be displayed and the VSI/HSI which will display the selected data. The CIS/MS panel is used to select either heading hold, altitude hold, or navigational information from the CISP. There are four major modes of operation of the CIS: Altitude hold mode, heading hold mode, go-around mode, and navigation mode. The CIS is off whenever none of the CIS modes have been selected or when the CIS navigation mode has been selected, but none of the required navigation data (Doppler, VOR/instrument landing system (ILS), or FM Home) has been selected on the pilot or copilot VSI/HSI Mode Selector panel.

Altitude Hold Mode

52. When the altitude hold mode is selected, the CISP uses barometric altitude and collective stick position data to provide collective commands to the VSI. The altitude hold mode can also be selected in any of the NAV modes except on an ILS approach when the ILS submode has been selected and the glideslope path has been captured or when go-around mode has been selected.

Heading Hold Mode

53. The heading hold mode may be manually selected by pressing the CIS/MS panel HDG switch, or it may be automatically selected as a navigation submode. In the manual HDG mode the CISP accepts the heading datum as set on the pilot or copilot HSI and the roll attitude signal from the attitude gyro and derives from them a limited cyclic roll command signal. This signal causes the vertical cyclic roll command bars on both VSI's to deflect. By steering the helicopter so that the roll command bar is kept centered, the pilot will maintain the selected heading within ±2 degrees. If any automatic heading hold mode is selected, the CISP provides a heading command which will intercept the desired NAV course. When the course is reached, the heading mode is automatically disengaged and the roll command bar is then controlled by NAV mode inputs.

Go-around Mode

54. The go-around mode is selected by depressing the GA switch on either the pilot or copilot cyclic grip. The CISP then provides pitch, roll and collective commands which will result in a 500 feet-per-minute climbout at 80 knots with zero bank angle. The pitch command is delayed 5 seconds. The CISP uses vertical gyro pitch and roll attitude, collective stick position, vertical velocity, and airspeed input signals to produce go-around commands. The go-around mode is disengaged by selecting another NAV mode on the pilot or copilot VSI/HSI mode select panel or by selecting another CIS mode on the CIS/MS panel.

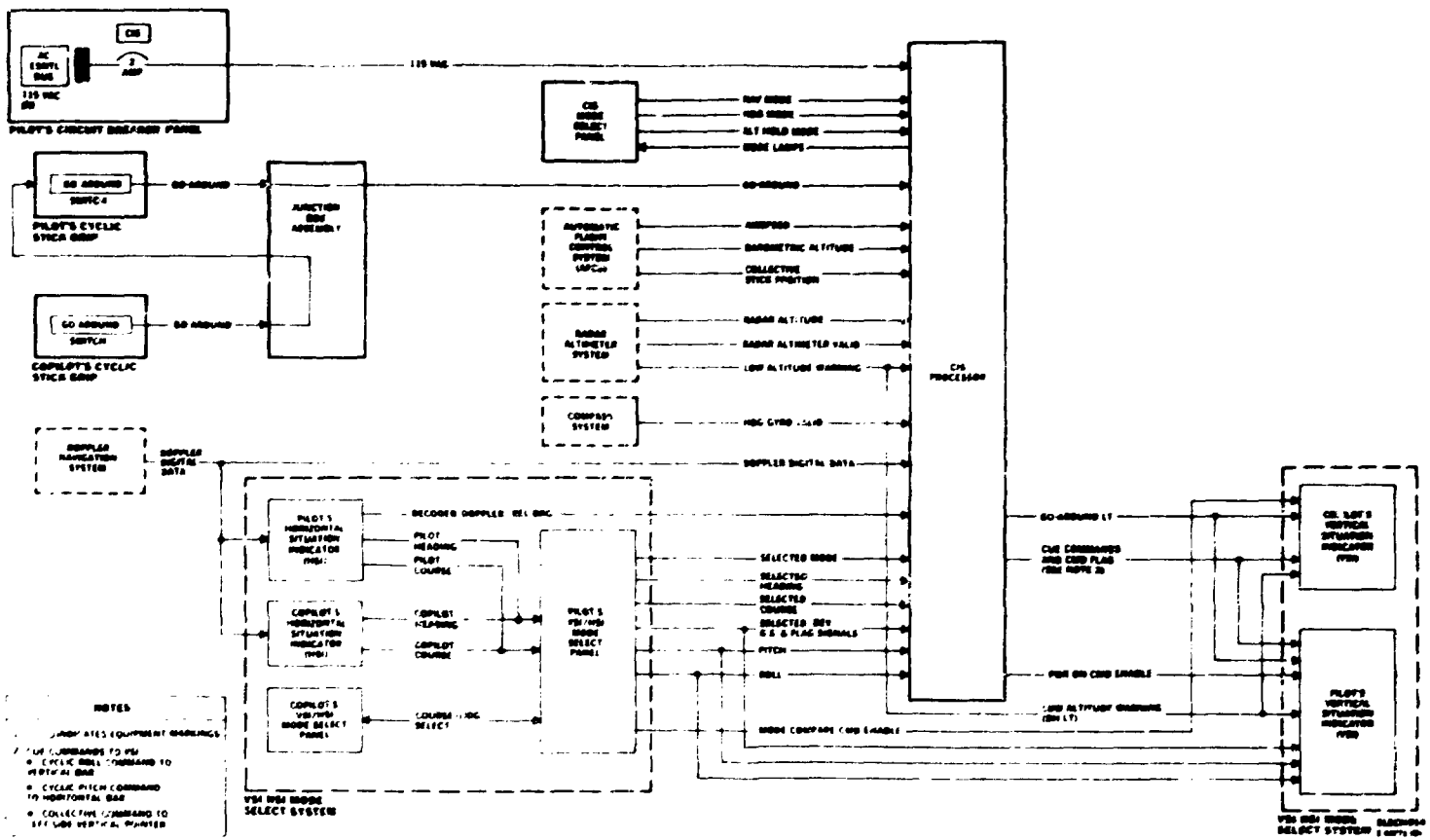


Figure 10. Command Instrument System (CIS) Block Diagram

Navigation Mode

55. The navigation mode has five submodes: Test, Doppler, FM Home, VOR, and ILS.

Test Submode:

a. The test submode is a self-test feature which causes the CISP to produce calibrated command outputs to the VSI's. The feature is a maintenance function and is enabled by pressing the CIS/MS panel NAV switch and the TEST button on the CISP front panel. The self-test indications are; roll command bar 0.5 inch right of center, pitch command bar 0.5 inch below center, collective command indicator 0.5 inch below center, and CMD flag in view.

Doppler Submode:

b. In the Doppler submode the CISP provides roll steering commands for straight line, wind corrected flight to a destination selected on the AN/ANS-128 Doppler Navigation System. The function is engaged by turning ON the Doppler Navigation System, pressing the pilot or copilot VSI/HSI mode select panel DPLR switch, and pressing the CIS/MS panel NAV switch. When the selected destination is reached, the NAV ON legend will go OFF and the heading hold mode will automatically be engaged.

FM Homing Submode:

c. In the FM homing submode the CISP uses vertical gyro roll attitude and FM homing signals to provide roll steering commands for the pilot to track to a FM station selected on the No. 1 VHF/FM system. The function is selected by selecting the HOMING mode on the No. 1 VHF/FM system, pressing the pilot VSI/HSI mode select panel FM HOME switch, and pressing the CIS/MS panel NAV switch. The CISP will detect station passage by lateral deviation rate and will automatically revert to the HDG mode until another mode is selected.

VOR Navigation Submode:

d. In the VOR navigation submode the CISP provides roll steering commands for the pilot to intercept and track a selected VOR radial. By selecting the desired VOR radial on the Omni Bearing Selector (OBS), pressing the pilot or copilot VSI/HSI mode select panel VOR/ILS switch, and pressing the CIS/MS panel NAV switch, the CISP will automatically provide roll steering commands to the final course intercept and subsequent course tracking information. When the NAV switch is pressed both the NAV ON and the HDG ON switch legends will be illuminated. The pilot must select the desired intercept angle and set the intercept heading in the HSI, thereafter, the CISP will provide roll steering commands to track the intercept heading. As the selected radial is intercepted the HDG ON legend will go out and radial tracking commands are displayed. Station passage is sensed by a movement of the lateral deviation indicator of 1 degree per second for more than 1 second. When these parameters are sensed the CISP reverts to heading hold, plus the last crab angle held, for ≈30 seconds after the last oscillations have been sensed. If a new radial is selected during the 30 second period, commands are given to parallel the new radial plus the last crab angle held. When the 30 second period was elapsed, commands will be given to intercept and track the selected radial.

ILS Navigation Submode:

e. In the ILS navigation submode, the CISP provides the same roll command signals as for VOR navigation and also provides an airspeed hold command to the pitch bar and an altitude hold command to the collective pointer. This submode is selected by tuning an ILS Localizer frequency, pressing the pilot or copilot VSI/HSI mode select panel VOR/ILS switch, and pressing the CIS/MS panel NAV switch. The CISP synchronizes to the airspeed and barometric altitude values received at the time of engagement and provides the VSI commands to maintain those values. Airspeed hold is limited to a range between 50 and 130 knots. Localizer course acquisition and tracking procedures are the same as for VOR navigation. The level-off function is also the same as for VOR navigation.

f. The ILS approach function is automatically engaged when the glideslope of the ILS is captured (not part of Localizer Back Course Approach) while following a localizer course. The CISP disengages the altitude hold function and provides a collective command signal to keep the aircraft on the glideslope. The approach function uses glideslope deviation and collective stick position signals to produce the collective pointer command; and airspeed signals from the air data transducer to produce the pitch bar command.

g. A level-off function is provided in the VOR/ILS navigation submode which will display collective pointer commands to level off at an altitude selected on the highest of the two LO SET warning indexes of the radar altimeters. The CISP uses collective stick position and the radar altimeter LO-SET warning index to produce collective command signals. The level-off function is automatically engaged during VOR or ILS navigation operation when the radar altitude drops below the LO SET warning index on either the pilot or copilot radar altimeter, which is higher. The function is automatically disengaged when the navigation and altitude hold functions are turned OFF or when an unreliable radar altimeter signal is detected.

h. The back course function is another submode of the ILS Approach mode. This function reverses the polarity of the localizer deviation indication and the pilot is provided cyclic roll commands, which, when properly followed, will allow the pilot to complete a localizer back course approach. The function is engaged by first selecting the ILS navigation function and then pressing the pilot or copilot VSI/HSI mode select panel BACK CRS switch. The localizer deviation signal is the only reference for this function.

BASIC AIRCRAFT INFORMATION

56. Principal dimensions and general data of the UH-60A helicopter are as follows:

Airframe

Length:

Maximum (rotor blades turning)	64 ft, 10 in.
Fuselage (nose to vertical tail)	) ft, 0.75 in.

Main rotor to tail rotor clearance	2.8 in.
Width:	
Main rotor blades turning	53 ft, 8 in.
Main landing gear	9ft, 8 in.
Height:	
Maximum (tail rotor blades turning)	16 ft, 10 in.
Main rotor clearance (ground to tip, rotor static against stops)	7 ft, 14 in.
Tail rotor clearance (ground to tip, rotor turning)	6 ft, 6 in.
Horizontal Stabilator:	
Span	172.6 in.
Chord - at root	44.0 in.
- at tip	30.5 in.
Aspect ratio	4.6
Airfoil section designation root to tip	NACA 0014
Sweep of leading edge, quarter chord	0 deg
Dihedral	0 deg
Range of travel (reference to fuselage water line)	39 deg trailing edge down to 9 deg trailing edge up
Taper ratio	1.87
Area (total)	45.0 sq ft
Vertical Tail	
Span	8 ft, 2 in.
Aspect ratio	1.92
Taper ratio	1.623
Sweep angle (1/4 chord line)	41 deg
Airfoil section designation and thickness (root to tip)	NACA 0021 to 65 percent with 7 deg trailing edge camber lower section

Incidence to fuselage reference line	0 deg
Area (total)	32.3 sq ft
Gross Weight	
Maximum alternate gross weight	20,250 pounds
Empty weight	Approximately 10,620 pounds
Primary Mission gross weight	16,260 pounds
Fuel capacity	364 gallons
Main Rotor	
Number of blades	4
Diameter	53 ft, 8 in.
Blade chord	1.73/1.75 ft
Blade twist	-18 deg (equiv)
Blade tip sweep	20 deg aft
Blade area (one blade)	46.7 sq ft
Geometric disc area (total)	2262 sq ft
Geometric solidity ratio (blade area/disc area)	0.0826
Airfoil section (root to tip) designation thickness (percent chord)	SC1095/SC1095R8 9.5 percent
Main rotor mast tilt (forward)	3 deg
Aspect ratio	15.4
Range of flapping	-6 to 25 deg
Blade droop stop angle (static) (flight)	-1/2 deg -6 deg
Tail Rotor	
Number of blades	4
Diameter	11 ft
Blade chord	0.81 ft

Blade twist (equiv linear)	-18 deg
Blade area (one blade)	4.46 sq ft
Geometric disc area (total)	95 sq ft
Geometric solidity ratio (blade area/disc area)	0.1875
Airfoil section (root to tip designation)	SC1095/SC1095R8
thickness (percent chord)	9.5 percent
Aspect ratio	6.79
Cant angle	20 deg

Main Rotor RPM

	Power On	Power Off
Minimum	234.7	232.1
Normal	245.0 to 260.5	232.1 to 270.8
Maximum	275.9	283.7
Design	257.9	—

Tail Rotor RPM

	Power On	Power Off
Minimum	1082.7	1070.8
Normal	1130.3 to 1201.7	1070.8 to 1249.3
Maximum	1273.1	1308.8
Design	1189.8	—

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

Rotational Speed Signals at 100 Percent

	RPM	Frequency, Hz
Main rotor, N_R	257.89	11,018.6
Power turbine, N_P	20,900	1393.3
Gas producer, N_g	44,700	2135.7

APPENDIX C. INSTRUMENTATION

GENERAL

1. The test instrumentation was installed, calibrated and maintained by the US Army Aviation Engineering Flight Activty (USAAEFA) personnel. 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 only for specific tests was installed when needed and is discussed in the section on special equipment. Data was obtained from calibrated instrumentation and displayed or recorded as indicated below.

Pilot Panel

Airspeed (boom)
Altitude (boom)
Altitude (radar)
Rate of climb*
Rotor speed (sensitive)
Engine torque* **
Turbine gas temperature* **
Power turbine speed (N_p)* **
Gas producer speed (N_g)* **
Control position
- Longitudinal
- Lateral
- Directional
- Collective

Horizontal stabilator position
Normal acceleration
Lateral acceleration (sensitive)
Angle of sideslip
Tether cable angle
Cable tension
Elliot low airspeed system
- Vertical
- Lateral
- Longitudinal

Copilot Panel

Event switch
Control fixtures
Airspeed*
Altitude*
Rotor speed*
Engine torque (both)*
Ballast cart control
Ballast cart position

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

Engineer Panel

Fuel remaining
APU fuel used
Instrumentation controls
Free air temperature
Time code display
Run number
Event switch

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

Digital (PCM) Data Parameters

Airspeed (boom)
Altitude (boom)
Altitude (radar - dual range)
Total air temperature
Rotor speed
Gas generator speed**
Power turbine speed**
Engine mass fuel flow**
Engine fuel used**
Lateral acceleration (sensitive)
Engine output shaft torque**
Turbine gas temperature**
Engine hydromechanical unit discharge pressure**
Engine compressor discharge pressure**
Engine stage 1 stator angle**
Engine power available spindle position**
Engine load demand spindle position**
Engine bleed air pressure**
Engine bleed air temperature**
Engine power turbine reference speed**
No. 1 Engine electrical control unit temperature
Engine starter supply air pressure**
APU fuel used
Main rotor shaft torque
Main rotor shaft temperature
Tether cable tension
Tether cable angle
    Longitudinal
    Lateral
Stabilator position
Movable ballast location
Control position
    Longitudinal cyclic
    Lateral cyclic
    Directional pedal
    Collective

**Both engines

SAS output position
- Longitudinal
- Lateral
- Directional

Mixer input position
- Longitudinal
- Lateral
- Directional

Primary servo position
- Lateral
- Forward
- Aft

Attitude
- Pitch
- Roll
- Yaw

Angular rate
- Pitch
- Roll
- Yaw

Linear acceleration
- Center of gravity normal
- Center of gravity lateral
- Center of gravity longitudinal

Tail rotor shaft torque
Tail rotor impressed pitch (Blade angle at 0.75 blade span)
Engine condition lever**
Elliot low airspeed system
- Longitudinal
- Vertical
- Lateral

Angle of sideslip
Angle of attack
Time of day
Run number
Pilot event
Engineer event

Analog (FM) Data Parameters

Vibration (accelerometers)
- Pilot seat vertical
- Pilot seat lateral
- Pilot seat longitudinal
- Center of gravity vertical
- Center of gravity lateral
- Center of gravity longitudinal
- Right instrument panel glare shield vertical
- Right instrument panel glare shield lateral
- Copilot instrument panel vertical

*Both engines.

Left aft cabin vertical
Right aft cabin vertical
Left forward cabin vertical
Pilot cyclic control vertical
Pilot pedal longitudinal

3. Provision was made for telemetry transmission of parameters.

AIRSPEED CALIBRATION

4. The standard ship's and test boom airspeed systems were calibrated during level flight, climb, and autorotation. The ground speed course was used to determine position error in level flight. A calibrated trailing bomb was utilized during climb and autorotation. The position error of the boom airspeed system is presented in figures 1 and 2.

SPECIAL EQUIPMENT

Low Airspeed System

5. The low airspeed system utilized during the vertical climb tests is manufactured by Marconi-Elliott Avionics Systems, LTD., Rochester, Kent, England. The system is designed to provide airspeed information in the longitudinal, lateral, and vertical axes of the helicopter. The system consists of a multiaxis omnidirectional probe which senses pitot and static pressure and angular data; an air data computer; longitudinal and lateral airspeed indicators; and a vertical speed indicator. A USAAEFA designed electronic interface box was used to take calculated longitudinal, lateral, and vertical airspeed data from the air data computer and display the longitudinal, lateral, and vertical airspeeds on indicators located on the pilot's instrument panel. The information was used to maintain zero horizontal airspeed during the vertical climb tests.

Weather Station

6. A portable weather station, consisting of an anemometer, sensitive temperature gage, and barometer, was used to record wind speed, wind direction, ambient temperature, and pressure altitude at selected heights up to 100 feet above ground level (AGL).

Digital Doppler Traffic Radar

7. The digital doppler traffic radar (speedgun) is a handheld battery operated device with a digital readout manufactured by CMI, Inc., Chanute, Kansas. The speedgun was used as a speed reference during the low-speed handling qualities tests.

Video Camera

8. A video camera was used as a means of reviewing test techniques because it could be replayed immediately after the test. No data parameters were recorded with this equipment.

FIGURE 1

MODEL SYSTEM AIRSPEED CALIBRATION IN LEVEL FLIGHT

UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
14,700	359.8(AFT)	0.0	1500	6.5	258

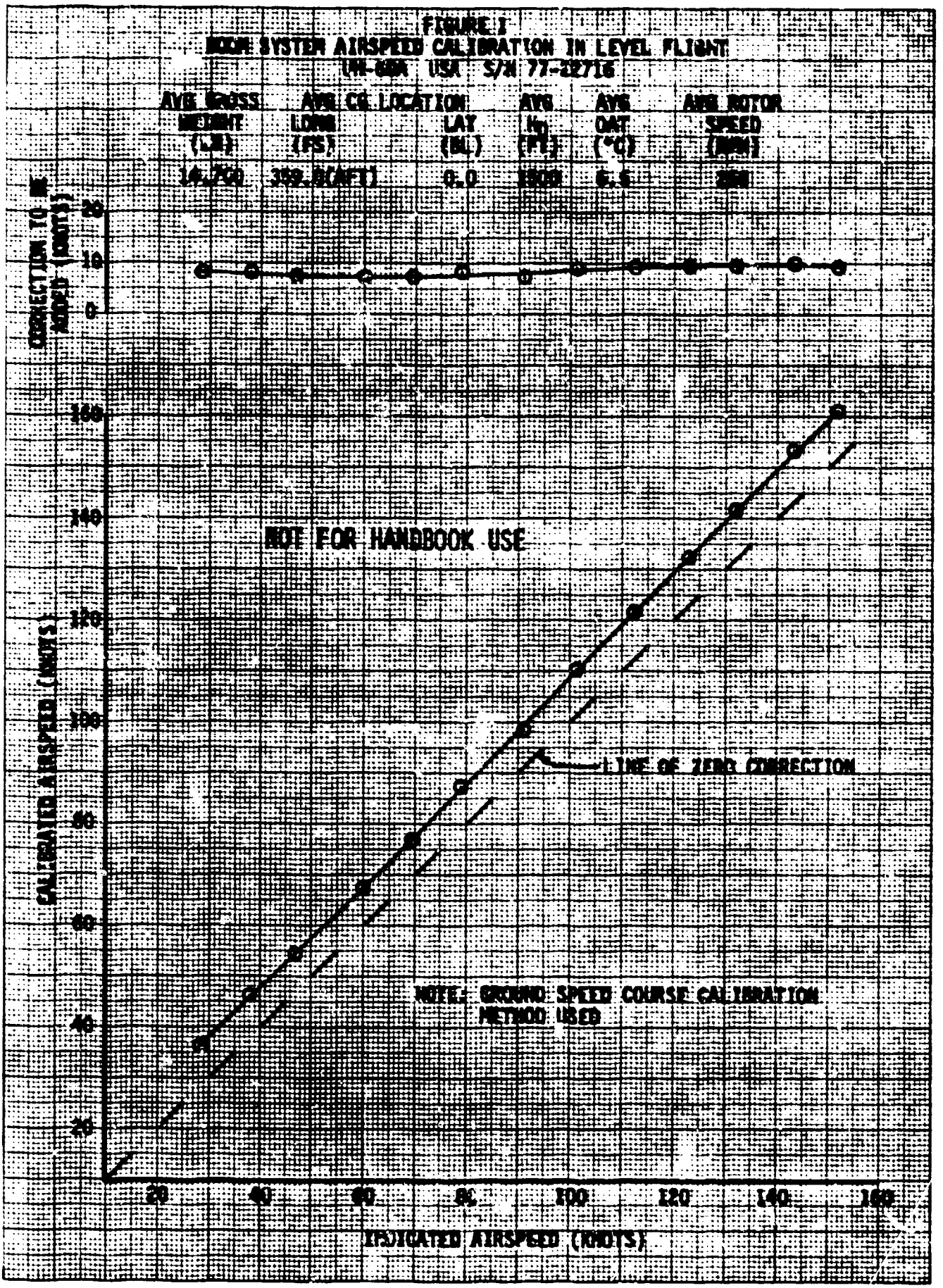
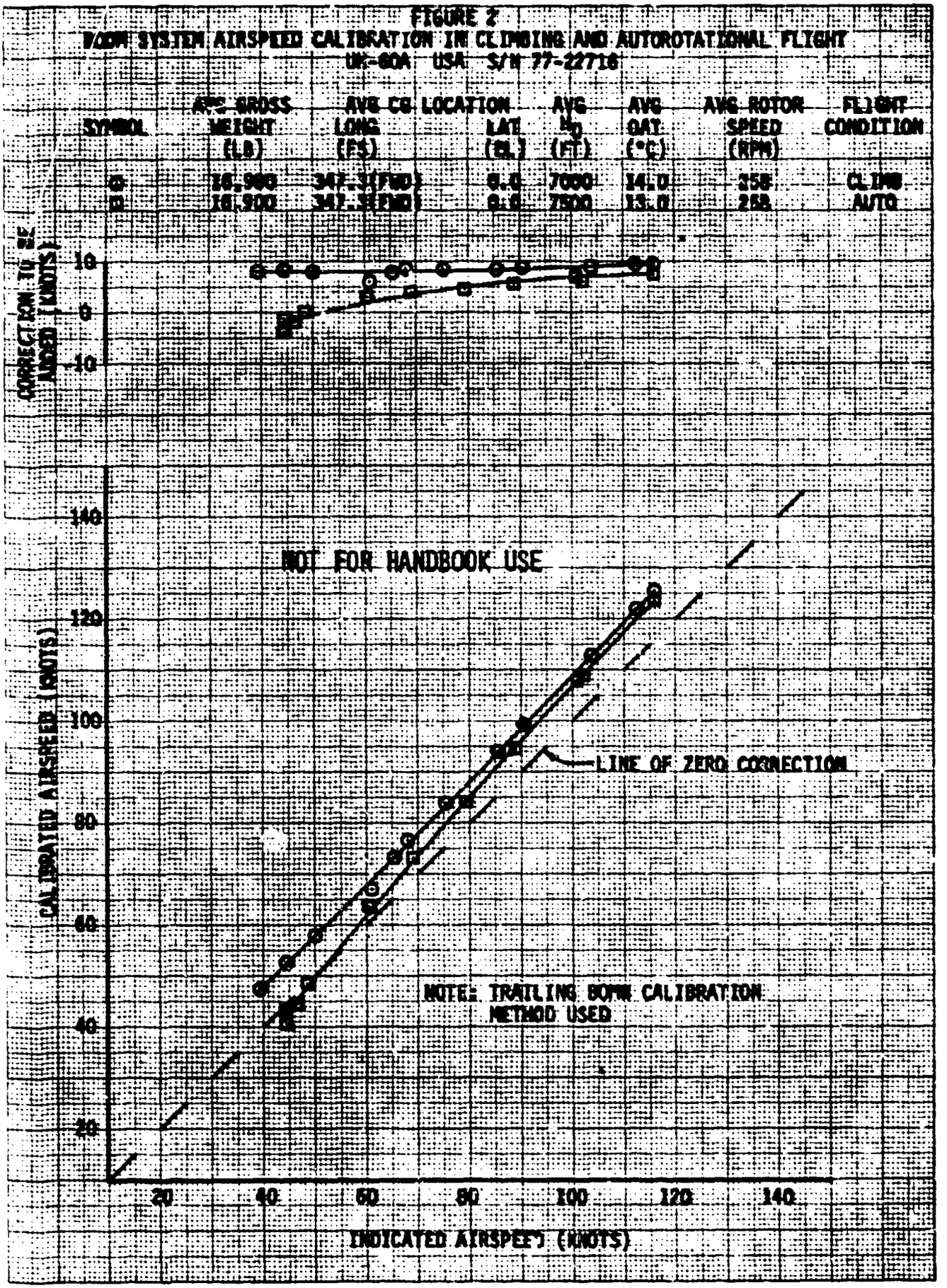


FIGURE 2

BOOM SYSTEM AIRSPEED CALIBRATION IN CLIMBING AND AUTOROTATIONAL FLIGHT

UH-60A USA S/N 77-22718

SYMBOL	APP. GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
⊙	16,900	347.3(FWD)	0.0	7000	14.0	258	CLIMB
⊡	16,900	347.3(FWD)	0.0	7500	13.0	258	AUTO



Load Cell

9. A calibrated load cell was incorporated with the ship's cargo hook and indicators were installed in the cockpit for displaying the cable tension as well as cable angle during the tethered hover tests. The indicators provided the pilot the amount of cable tension being pulled on the cable and the longitudinal and lateral displacement of the helicopter over the tie-down point on the ground.

Ground Pace Vehicles

10. Pace vehicle speedometers were calibrated with the digital doppler traffic radar (speedgun). The pace vehicles were used in conjunction with the speedgun to establish precise airspeed during the low airspeed handling qualities tests.

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. Performance data were obtained using the basic methods described in Army Materiel Command Pamphlet AMCP 706-204 (ref 7, app A). Performance testing was conducted in coordinated (ball-centered) flight. Handling qualities data were evaluated using standard test methods described in Naval Air Test Center Flight Test Manual FTM No. 101 (ref 8).

AIRCRAFT WEIGHT AND BALANCE

2. The aircraft was weighed in the instrumented configuration with full oil and all fuel drained prior to the start of the A&FC program. The initial weight of the aircraft was 11,820 pounds with the longitudinal center of gravity (cg) located at FS 354.0 with the cg of the empty ballast cart located at FS 301. The aircraft was periodically weighed during the program as items necessary for various tests were added or deleted. The aircraft was also weighed with the IR suppressor, XM-130 chaff dispenser, and a mock-up of the AN/ALQ-144 IR jammer installed. This increased the gross weight of the aircraft by 330 pounds. The fuel cells and an external sight gage were 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 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 was compared with the sight gage readings. Aircraft cg was controlled by a moveable ballast system which was manually positioned to maintain a constant cg while fuel was burned. The moveable 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 generalized through the use of non-dimensional coefficients as follows using the 1968 US Standard Atmosphere:

a. Coefficient of Power (C_P):

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

b. Coefficient of Thrust (C_T):

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

c. Advance Ratio (μ):

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

Where:

SHP = Engine output shaft horsepower (both)

ρ = Ambient air density (lb-sec^2/ft^4)

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 = True airspeed (kt) = $\dfrac{V_E}{1.6878\sqrt{\rho/\rho_o}}$

1.6878 = Conversion factor (ft/sec-kt)

ρ_o = 0.0023769 (lb-sec^2/ft^4)

V_E = Equivalent airspeed (ft/sec) =

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

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

Q_c = Dynamic pressure (in.-Hg)

P_a = Ambient air pressure (in.-Hg)

100% rotor speed = 257.9 rpm

ΩR = 724.69

$(\Omega R)^2$ = 525168.15

$(\Omega R)^3$ = 380581411.2

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 mechanically displays 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 was calibrated in a test cell by the engine manufacturer. The output from the engine torque sensor was recorded on the on-board data recording system. The output SHP was determined from the engine's output shaft torque and rotational speed by the following equation.

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

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.

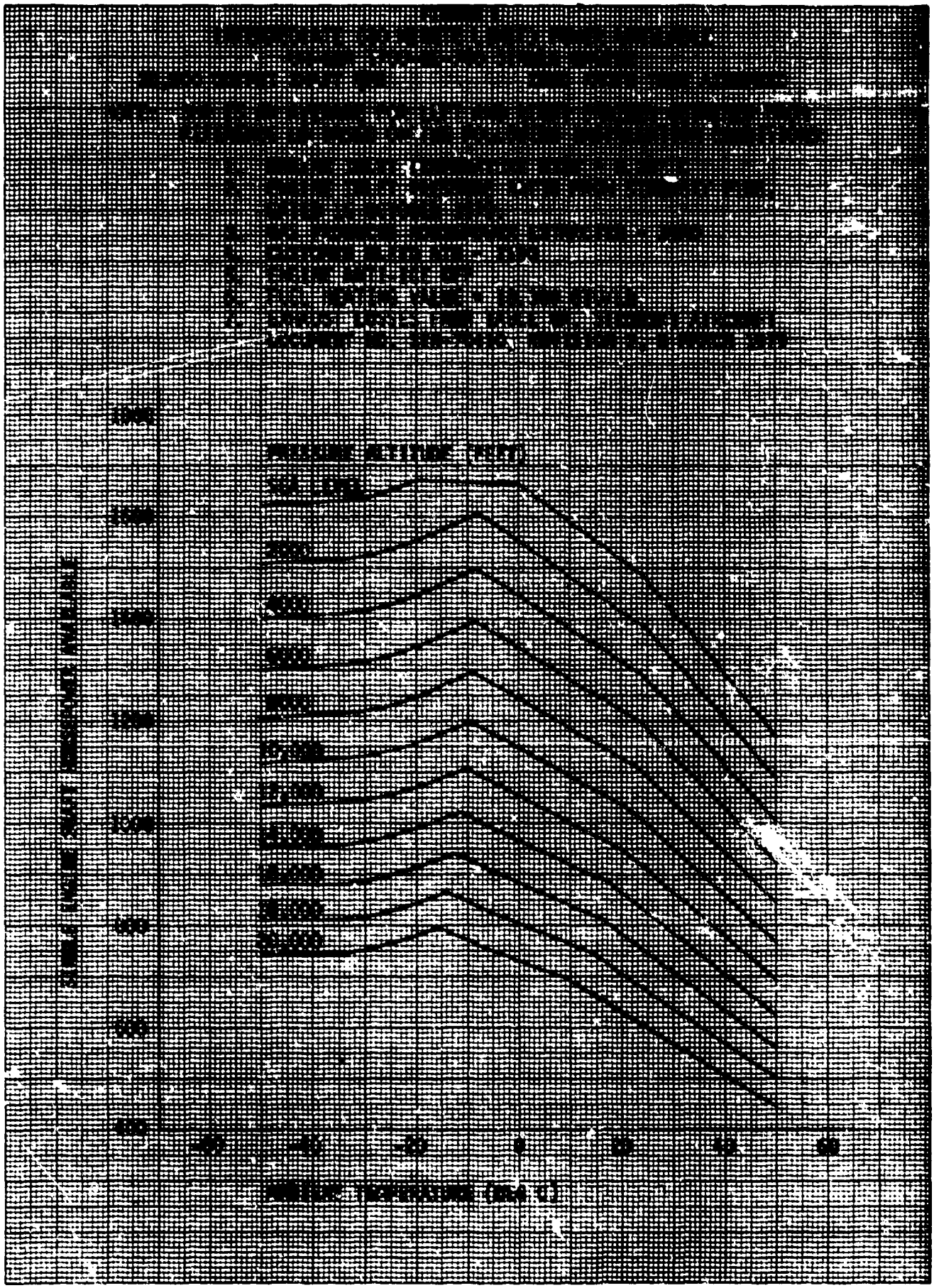
Shaft Horsepower Available

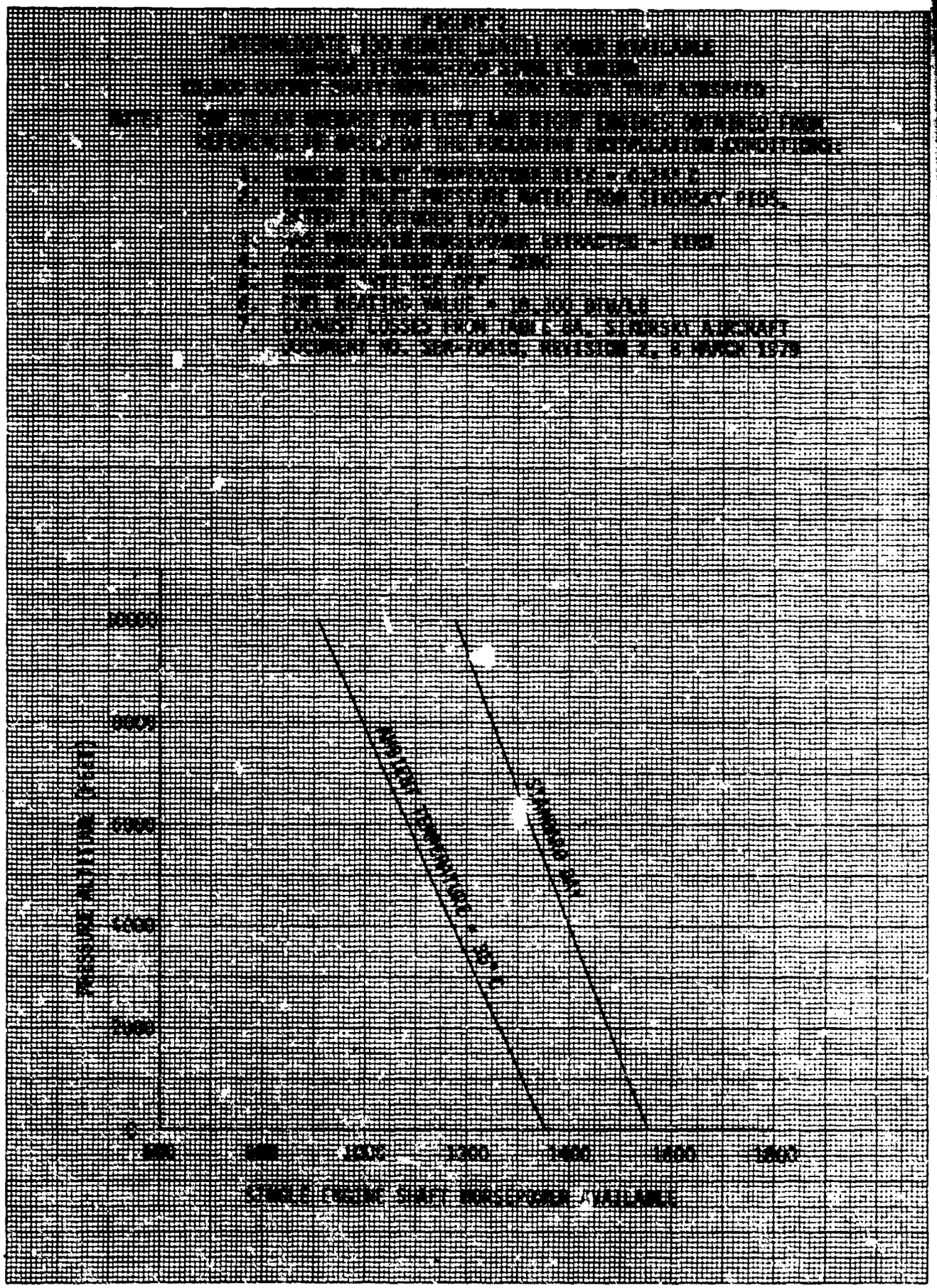
5. Shaft horsepower available for the T700-GE-700 engine installed in the UH-60A was obtained from data received from AVRADCOM (ref 9, app A). This data was calculated using the General Electric engine deck number 80024, dated 26 February 1981 with a power turbine shaft speed of 20,900 rpm. The installation losses used were based on an arbitrary 0.25 degree C for 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 PIDS, and an inlet temperature rise in forward flight assuming an adiabatic rise referenced to a zero degree rise in a hover. The single engine shaft horsepower available is assumed to be an average of the left and right engines and is presented in figures 1 through 3, appendix D. These figures were used to determine the PIDS commitment for hover, takeoff, vertical climb and level flight performance.

Hover

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 increasing gross weight. Free-flight hover tests consisted of stabilizing the helicopter at a desired height using the radar altimeter as a height reference. Atmospheric pressure, temperature, and wind velocity were recorded from a ground weather station. All hovering data were reduced to nondimensional parameters of C_P and C_T (equations 1 and 2, respectively), and grouped according to wheel height and average density altitude. A line was faired through each set of data. Summary




2000
4000
6000
8000
1000
1200
1400
1600
1800

hovering performance was then calculated from these nondimensional plots using the power available from figure 2, appendix D.

Tail Rotor Performance:

7. Nondimensional tail rotor performance and directional control position were used to determine tail rotor horsepower and directional control margins as a function of wheel height. Terms in equations 1 and 2 which apply to the main rotor were replaced by tail rotor parameters for nondimensionalized tail rotor performance. Antitorque system output torque was measured at the input shaft to the intermediate gearbox. The horsepower loss due to the intermediate and 90-degree gearboxes were determined by Sikorsky Aircraft and incorporated by a correction factor (0.988036) applied to tail rotor drive shaft horsepower to calculate tail rotor shaft horsepower. The terms redefined are as follows.

SHP = Tail rotor shaft horsepower (SHP_{TR})

A = Tail rotor disc area = 95.033 ft^2

Ω = Tail rotor angular velocity (radians/sec)

R = Tail rotor radius = 5.50 ft

GW = Tail rotor thrust (lb) (THRUST)

Tail rotor shaft horsepower was determined from the following equation.

$$SHP_{TR} = \left[\frac{\left(\frac{N_P}{15.9583}\right) \quad (Q_{TRD})}{5252.113} \right] \quad 0.988036 \qquad (5)$$

Where:

Q_{TRD} = Torque at input shaft of intermediate gearbox

15.9583 = Gear ratio of engine to tail rotor drive shaft

8. The component of tail rotor thrust necessary for stabilized hover was determined by making two assumptions. These assumptions were necessary since tail rotor thrust could not be measured directly during the evaluation. The first assumption was that all directional moments to maintain stabilized hover were generated by the antitorque tail rotor. This assumption neglected any possible restoring directional moments that could be derived from rotor downwash and recirculating airflow over the fuselage, tail boom section, and empennage. The second assumption was that the temperature of the air passing through the tail rotor was not influenced by the engine exhaust gases. The restoring component of tail rotor thrust was determined from the following equation.

$$THRUST = \frac{QMR}{l_T} \qquad (6)$$

Where:

QMR = Main rotor shaft torque (ft-lb)

l_T = Perpendicular distance between center lines of main and tail rotor shafts = 32.567 ft

Climb

Vertical Climb Performance:

9. The vertical climb technique used was to stabilize in a 100-foot OGE hover based on the radar altimeter and then increase engine power by a predetermined increment of engine torque. Various increments of engine torque up to the engine limits were used. An Elliott low-airspeed system was used to provide cues of longitudinal and lateral translation. Each vertical climb was flown at a predetermined C_T and referred rotor speed, and during the climb rotor speed held constant. Ballast was added as fuel burned off or temperature varied. Tests were initiated at 100 feet in ambient wind conditions of 3 knots or less.

10. The climb rates were measured after the aircraft was stabilized in unaccelerating vertical climbing flight by differentiating the output of the radar altimeter with respect to time. Vertical climb performance was determined nondimensionally in terms of vertical velocity ratio and generalized power coefficient defined as follows.

a. Vertical velocity ratio (VVr).

$$VVr = \frac{Vv}{V_{tip}\left[\frac{C_T}{2}\right]^{1/2}} \tag{7}$$

b. Generalized excess power coefficient (variation from hover) (ΔC_P gen).

$$\Delta C_P \text{ gen} = \frac{C_{P_C} - C_{P_H}}{\left[\frac{C_T^3}{2}\right]^{1/2}} \tag{8}$$

Where:

Vv = Vertical velocity (ft/sec)

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

C_{P_C} = Coefficient of power for climb

C_{P_H} = Coefficient of power for hover

The summary vertical climb performance was determined from the nondimensional plot using the OGE hover performance, figure 3, appendix E, and the power available, figure 2, appendix D.

Forward Flight Climb Performance:

11. Two series of climbs were conducted to determine the power and weight correction factors (K_P and K_W). A constant rotor speed and predetermined power and airspeed schedules were used. The climb airspeed schedule was determined by the airspeed corresponding to the minimum power required for level flight based on preliminary data. To obtain K_P data, one series of climbs was flown at a constant aim gross weight from altitudes near sea level up to 15,000 feet at various power settings. This series of climbs was corrected to a constant gross weight. For K_W data, the other series of climbs was flown with dual-engine IRP at various gross weights, from altitudes near sea level to each respective climb service ceiling or 15,000 feet, whichever was lower. This series of climbs was corrected for deviations from the aim power schedule. Corrected test results of rate of climb versus SHP and gross weight are presented in figures 13 and 14, appendix E. A constant value of 0.76 was determined for K_P, while K_W was found to vary as a function of gross weight throughout the altitude range tested. Power and weight factors were determined using the following equations.

a. Power correction factor (K_P):

$$K_P = \left[\frac{\Delta R/C}{\Delta SHP}\right]\left[\frac{GW_t}{33{,}000}\right] \tag{9}$$

b. Weight correction factor (K_W):

$$K_W = \frac{(R/C_2 - R/C_1)}{(SHP)\,(33{,}000)}\left[\frac{(GW_t)\,(GW_2)}{GW_1 - GW_2}\right] \tag{10}$$

Where:

$\Delta R/C$ / ΔSHP = Change in rate of climb for a corresponding change in SHP, from R/C versus SHP curve, figure 13, appendix E.

GW_t = Test gross weight

$R/C_1/GW_1$ = Heavy gross weight and corresponding rate of climb from R/C versus GW curve, figure 14.

$R/C_2/GW_2$ = Light gross weight and corresponding rate of climb from R/C versus GW curve, figure 14.

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

12. Power corrections were applied for variations in airspeed from the climb airspeed schedule. Any deviations from this minimum power airspeed were corrected by the following equation.

$$\Delta R/C_P = \frac{(K_P)\,(\Delta SHP)\,(33{,}000)}{GW_t} \tag{11}$$

Where:

ΔSHP = Difference in level flight power required at test conditions between the test airspeed and climb schedule airspeed.

13. Single engine climb performance was determined by the following procedure to check the PIDS requirement. The shp required for level flight at the mission gross weight was calculated from the level flight performance tests conducted at a referred rotor speed of 258 rpm. Power available was obtained from figure 2, appendix D, for the 35° C and 5000 H_p static conditions. The change in power due to ram effects was determined from figure 3 at 83.7 KTAS, at 35° C and 4000 H_p conditions. The forward speed increased the shp available by 16 shp and this increase was added to the shp available for static conditions at 35° C and 5000 H_p conditions. The power available and power required were used to determine the single engine climb performance using equation 11.

Level Flight Performance

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

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

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

$$\mu = \frac{V_T\,(16.12)}{R\sqrt{\theta}\left[\frac{N_R}{\sqrt{\theta}}\right]} \tag{14}$$

Changes in horsepower due to changes in flat plate area were determined from the following equation.

$$\Delta SHP = \frac{(\Delta F_e)(\sigma)(V_T^3)}{96254} \quad (15)$$

Where:

δ = Pressure ratio = $\frac{P_s}{P_{s_o}}$

P_{s_o} = 29.92126 in.-Hg

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

OAT = Ambient air temperature (° C)

N_R = Main rotor speed (rev/min)

$\sigma = \delta/\theta$

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)

ΔF_e = Change in equivalent flat plate area (ft^2)

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

15. Each speed power was flown in ball centered flight by reference to a sensitive lateral accelerometer at a predetermined coefficient of thrust (C_T) and referred rotor speed ($N_R/\sqrt{\theta}$). To maintain the ratio of gross weight to pressure ratio (W/δ) 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.

$$SHP_{R/C} = -\frac{(R/C_{TL})(GW)}{33{,}000(K_p)} \quad (16)$$

$$SHP_{ACCEL} = -1.6098 \times 10^{-4} \left(\frac{\Delta V}{\Delta t}\right)(V_T)(GW) \quad (17)$$

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 of $\frac{\Delta H_p}{\Delta t}$ (°C)

$K_p = 0.76$

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

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

Reductions in power required were made for the installation of test instrumentaion. A power loss of 1.82 shp was applied for electrical operation of the instrumentation. The effects of instrumentation drag were determined by equation 15, where ΔF_e = 0.833 ft^2 of external instrumentation and $\Delta SHP_{INSTR\ DRAG}$ = ΔSHP. Power required for level flight at the test day conditions was determined using the following equation.

$$SHP_t = SHP + SHP_{R/C} + SHP_{ACCEL} - \Delta SHP_{INSTR\ DRAG} - 1.82 \qquad (18)$$

16. Test-day level flight data was corrected to standard-day conditions by the following equations.

$$SHP_s = SHP_t \frac{(\delta_s \sqrt{\theta_s}) \left[\frac{N_R}{\sqrt{\theta}}\right]_s^3}{(\delta_t \sqrt{\theta_t}) \left[\frac{N_R}{\sqrt{\theta}}\right]_t^3} \qquad (19)$$

$$V_{T_s} = V_{T_t} \left[\frac{\frac{N_R}{\sqrt{\theta}}}{\frac{N_R}{\sqrt{\theta}}}\right]_t^s \qquad (20)$$

Where:

subscript t = Test day

subscript s = standard day

Test data corrected for rate of climb, acceleration, instrumentation installation, and corrected to standard altitude and ambient temperature are presented in figures 47 through 76, appendix E.

17. Conversion of ball centered test data to a zero sideslip trim condition was made for the purpose of data analysis. Flights were flown at different constant C_T's and referred rotor speeds varying aircraft angle of sideslip at various airspeeds. A unique curve (fig. 46, app E) of change in equivalent flat plate area as a function

of sideslip angle was derived independent of C_T and $N_R/\sqrt{\theta}$. Changes in power required due to inherent sideslip were made utilizing this unique curve, and equation 15, to determine differences in power required between the sideslip angle measured in ball centered flight and zero sideslip trim conditions. Power required data for level flight used in calculation of C_P for this analysis (SHP_{C_P}) was finally determined by the following equation.

$$SHP_{C_P} = SHP_t + \Delta SHP \tag{21}$$

Where:

ΔSHP = Change in SHP from ball centered to zero sideslip

18. Data analysis was accomplished by plotting power coefficient (C_P) versus advance ratio (μ) for each test average C_T and $N_R/\sqrt{\theta}$. These curves were then cross-faired as C_P versus C_T for an initial determination of what effect $N_R/\sqrt{\theta}$ had at a given μ throughout the level flight envelope. These curves were then cross faired as C_P versus $N_R/\sqrt{\theta}$ at selected increments of C_T for all previous μ increments to further refine trends between values of μ and $N_R/\sqrt{\theta}$. These curves were subsequently faired into individual carpet plots (C_T versus C_P for a constant μ value) at each $N_R/\sqrt{\theta}$ at the aim test conditions (average $N_R/\sqrt{\theta}$ for 245 rpm to prevent extrapolation beyond data) to finalize all curves (figs. 31 through 42, app E). The reduction of these carpet plots into related families of curves (C_P versus $N_R/\sqrt{\theta}$ for constant C_T at increments of μ) allows determination of the power required as a function of airspeed for any value of C_T and $N_R/\sqrt{\theta}$ (figs. 15 through 30).

19. Modification of power required from zero sideslip to ball centered flight conditions was accomplished by developing a carpet plot of inherent sideslip, attained in conjunction with level flight performance, as a function of C_T for increments of μ (determined to be independent of $N_R/\sqrt{\theta}$). Utilizing this family of curves for obtaining the sideslip angle associated with ball centered flight for a specified C_T and μ, ΔF_e was obtained from figure 46 for this angle of sideslip, and ΔC_P for converting carpet plotted power required was obtained as follows.

$$\Delta C_P = \frac{\Delta F_e \, \mu^3}{2A} \tag{22}$$

Where:

ΔC_P = Change in coefficient of power

The power coefficient for ball centered flight, at any desired C_T and μ, was then obtained by the following equation.

$$C_{P_{BALL\ CENTER}} = C_{P_{ZERO\ SIDESLIP}} + \Delta C_P \tag{23}$$

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

$$SHP_{REF} = \frac{SHP_t}{\delta \theta^{0.5}} \tag{24}$$

$$W_{F_{REF}} = \frac{W_{F_t}}{\delta \theta^{0.55}} \tag{25}$$

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.

$$W_{F_s} = W_{F_t} + \Delta W_F \tag{26}$$

Where:

ΔW_F = Change in fuel flow between SHP_t and SHP_s

The following equation was used for determination of specific range.

$$SR = \frac{V_{T_s}}{W_{F_s}} \tag{27}$$

21. Changes in the equivalent flat plate area due to change in aircraft configuration and cg were calculated from equation 15 solved for ΔF_e. Where ΔSHP is the difference in SHP as derived from the nondimensional plots at the normal utility forward cg configuration and the SHP for the configuration or cg desired. Inherent sideslip data for different configurations and cg locations tested was within the spread of data accumulated in the normal utility forward cg configuration and, therefore, the carpet plot of paragraph 18 for conversion to ball center data was considered applicable.

Autorotational Descent Performance

22. Autorotational descent performance data were acquired at various stabilized airspeeds with constant rotor speed and a range of stabilized rotor speeds at a constant airspeed. The tapeline rates of descent were calculated by the following equation.

$$R/D_{TL} = \left(\frac{\Delta H_P}{\Delta t}\right)\left(\frac{OAT + 273.15}{OAT_s + 273.15}\right) \tag{28}$$

HANDLING QUALITIES

General

23. Conventional test techniques were used during the conduct of the handling qualities tests. All tests were conducted during ball centered flight except for the maneuvering stability test, which was conducted during zero sideslip flight. Detailed descriptions of all test techniques are contained in reference 8, appendix A.

VIBRATION

24. Vibration data were analyzed using a Spectral Dynamics Model SD301B real time spectral analyzer. The analyzer converted the data from the time domain (acceleration as a function of time) to the frequency domain (acceleration as a function of frequency). The data were analyzed using a frequency range of zero to 100 Hz and frequency resolution of 0.5 Hz. In order to minimize random variation in acceleration amplitude, the data were averaged over a 20-second time interval in level flight and a 5-second time interval in maneuvering flight using a Spectral Dynamics Model SD302B ensemble averager.

25. The instrument panel accelerometers were attached to the extreme left and right portion of the instrument panel glare shield. Their location may not register the actual vibration level of the panel because of the shields flexibility. Therefore no vibration data for the instrument panel was presented in this report.

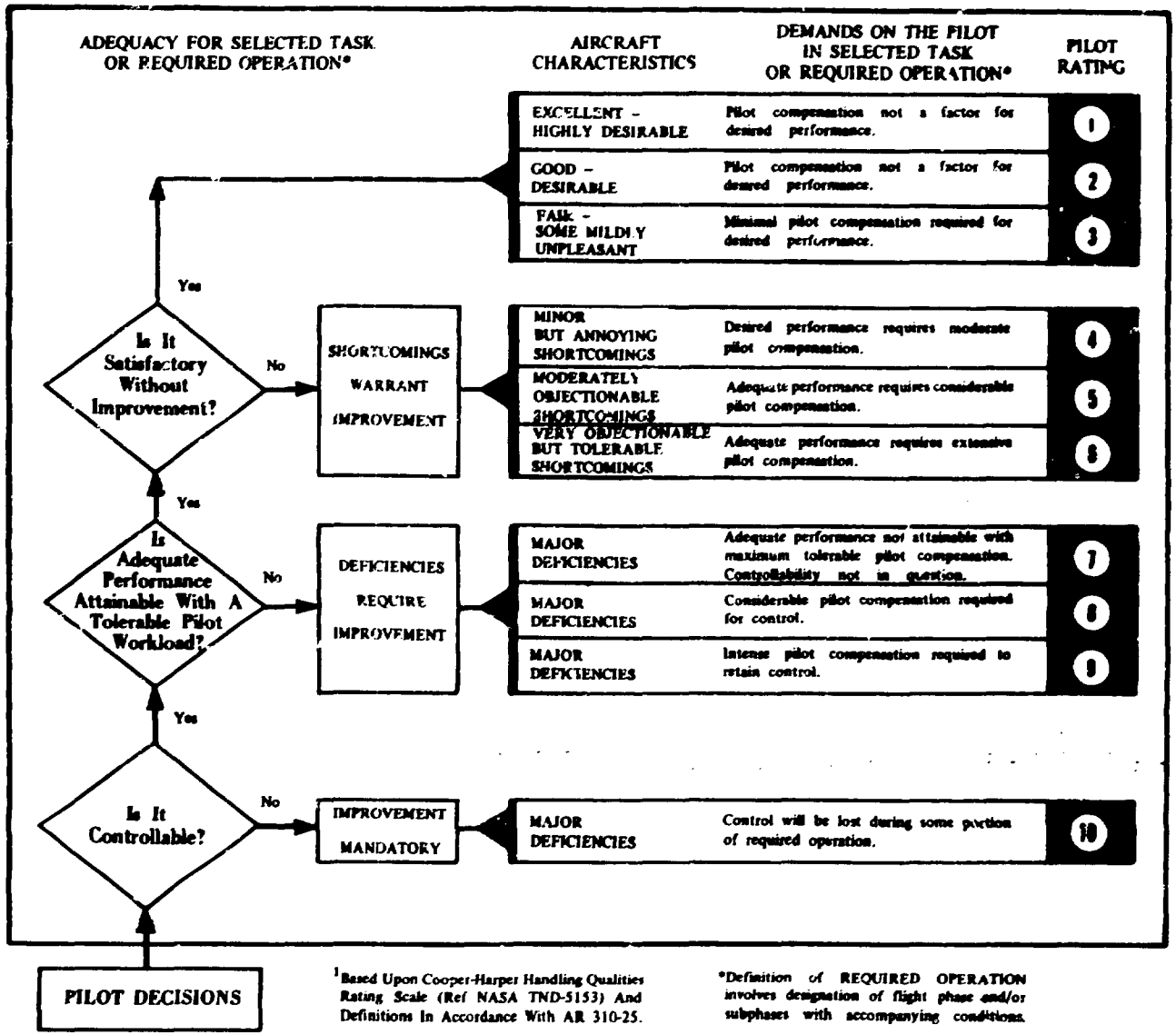
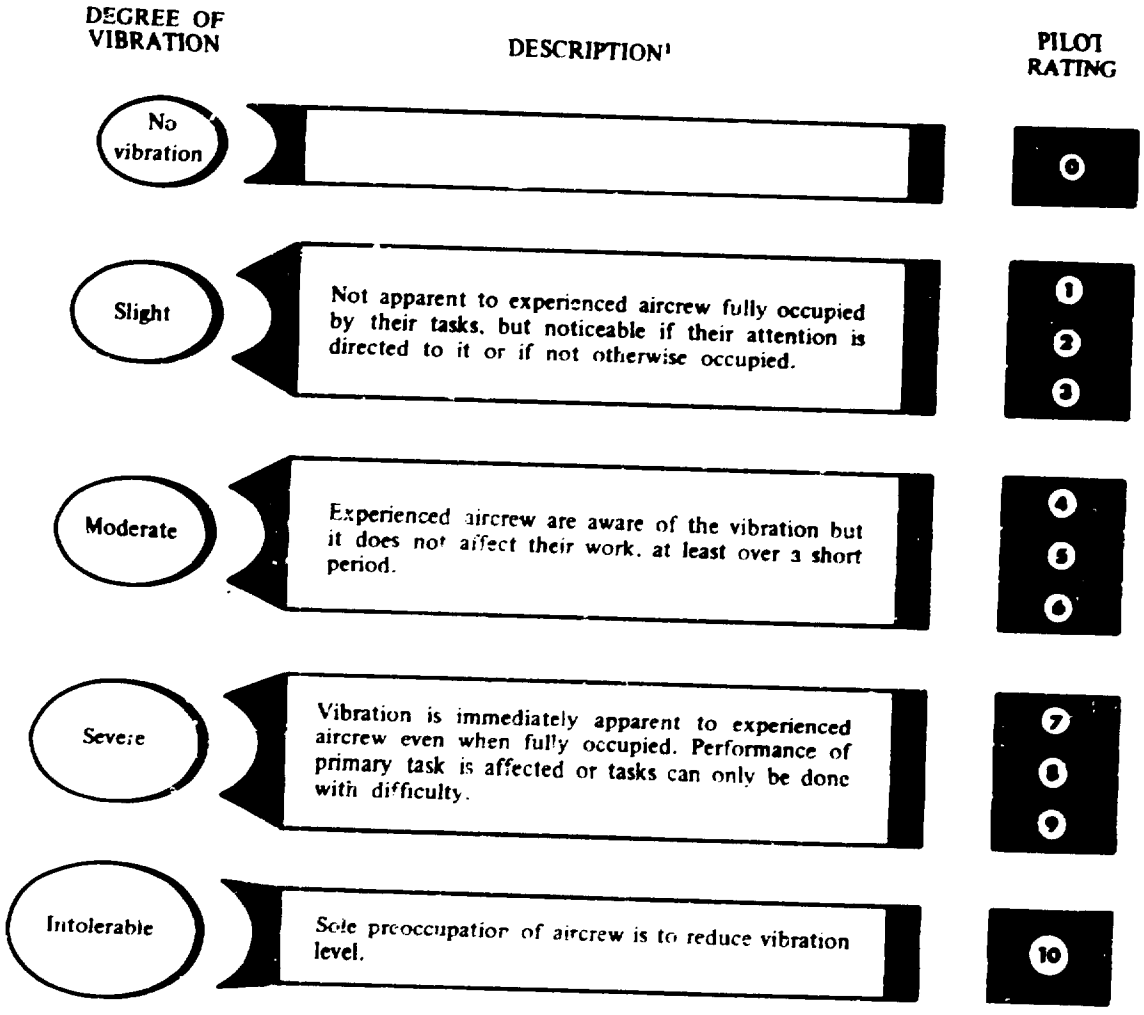


Figure 4. Handling Qualities Rating Scale



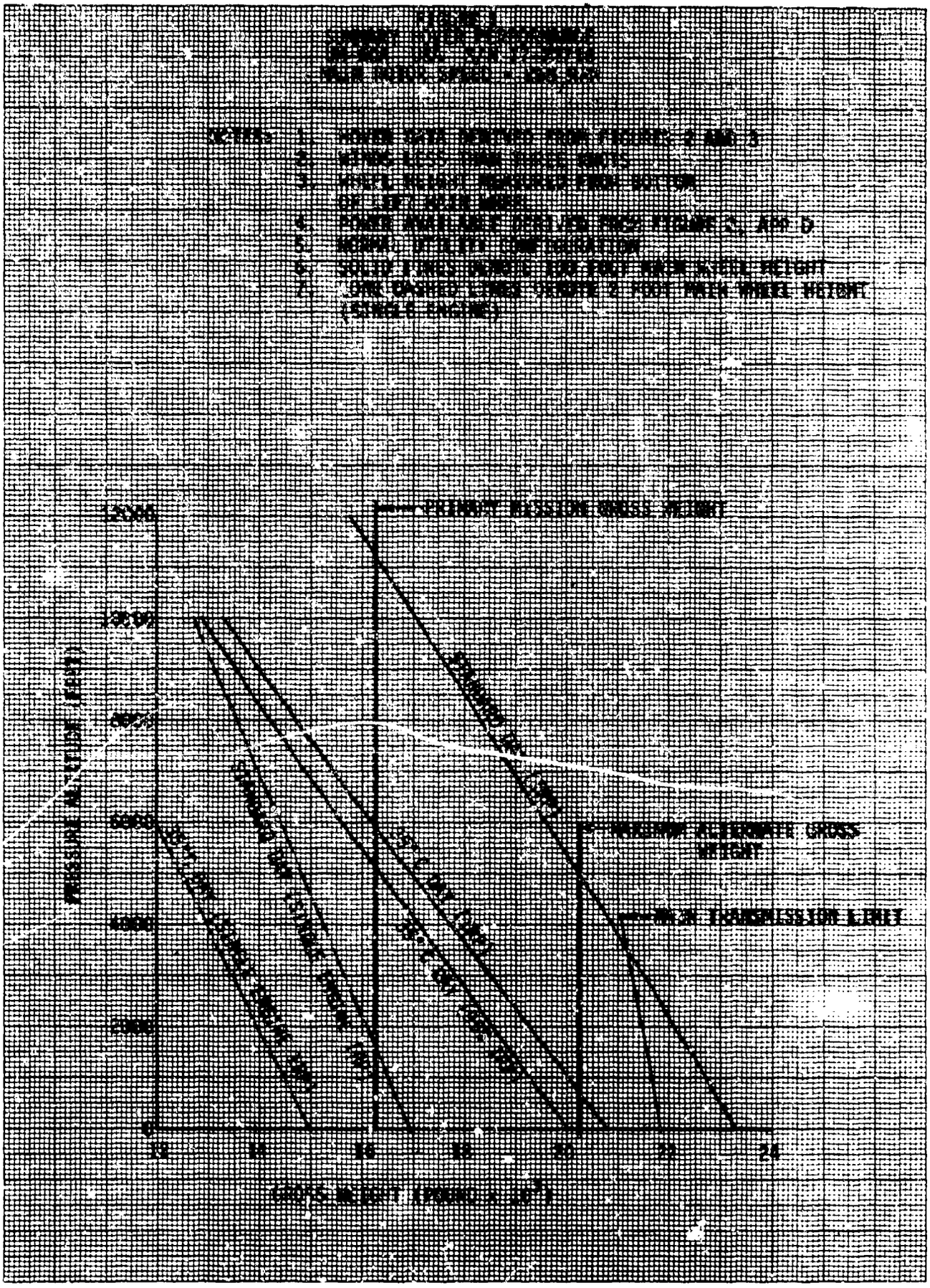
[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 5. Vibration Rating Scale

APPENDIX E. TEST DATA

INDEX

Figure	Figure Number
Hover Performance	1 through 10
Vertical Climb Performance	11 and 12
Forward Flight Climb Performance	13 and 14
Level Flight Performance	15 through 76
Autorotational Descent Performance	77 and 78
Control System Characteristics	79 through 84
Control Positions in Trimmed Forward Flight	85 through 90
Collective-fixed Static Longitudinal Stability	91 and 92
Maneuvering Stability	93 and 94
Controllability	95 through 99
Takeoff Characteristics	100 and 101
Low-speed Flight Characteristics	102 through 110
Engine Starts	111 and 112
Vibration Characteristics	113 through 124
Airspeed System	125 and 126


PERMIT MISSION GROSS WEIGHT
MAXIMUM ALTERNATE GROSS WEIGHT
M.A.M. TRANSMISSION LIMIT
GROSS WEIGHT (POUNDS x 10³)
12
16
20
24

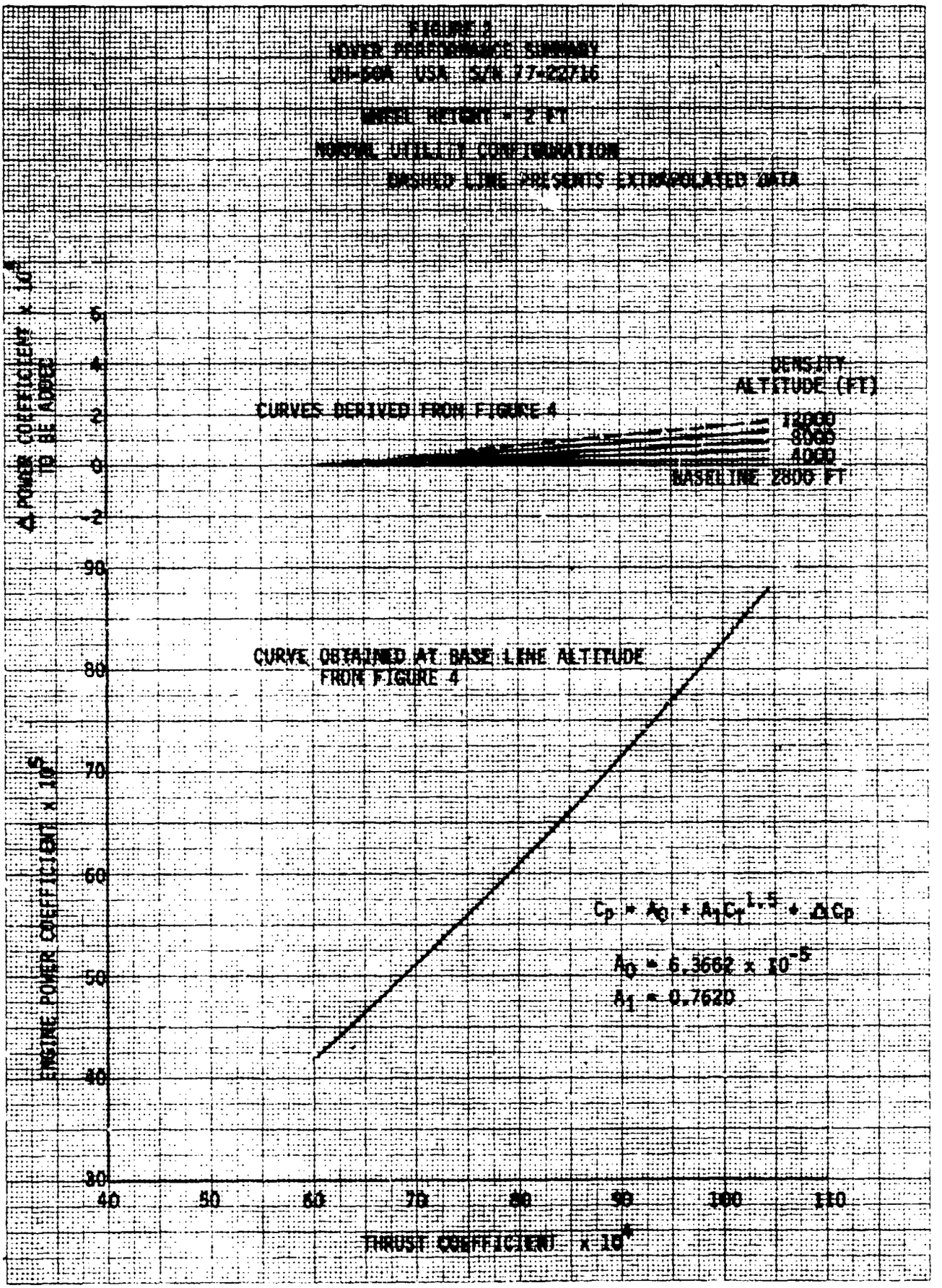
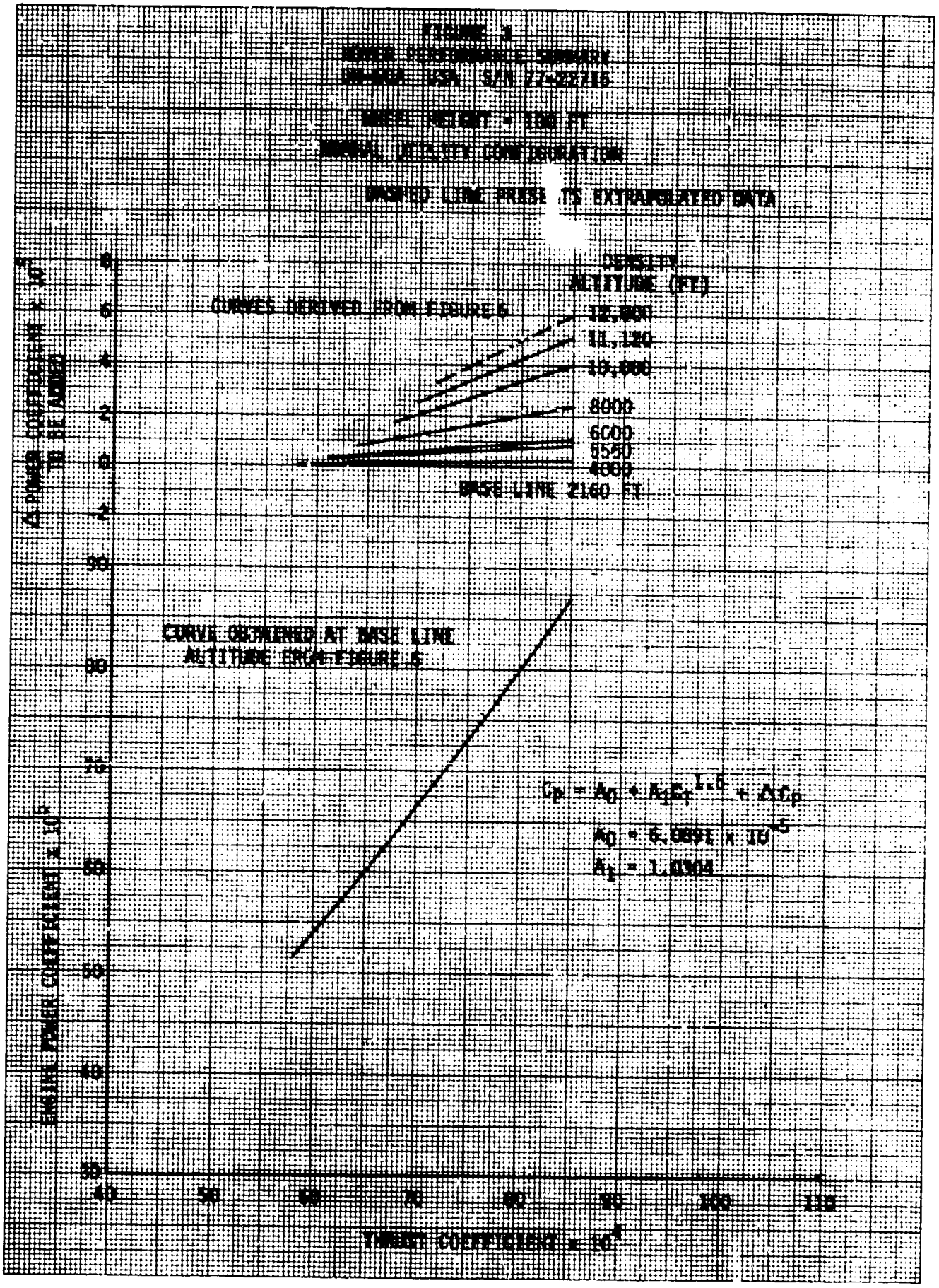


FIGURE 3
HOVER PERFORMANCE SUMMARY
UH-60A USA S/N 77-22716
WHEEL HEIGHT = 2 FT
NORMAL UTILITY CONFIGURATION
DASHED LINE REPRESENTS EXTRAPOLATED DATA
ΔPOWER COEFFICIENT x 10⁴ TO BE ADDED
6
4
2
0
-2
DENSITY ALTITUDE (FT)
CURVES DERIVED FROM FIGURE 4
12000
8000
4000
BASELINE 2800 FT
90
80
70
60
50
40
30
ENGINE POWER COEFFICIENT x 10⁵
CURVE OBTAINED AT BASE LINE ALTITUDE FROM FIGURE 4
$C_P = A_0 + A_1 C_T^{1.5} + \Delta C_P$
$A_0 = 6.3662 \times 10^{-5}$
$A_1 = 0.7620$
40
50
60
70
80
90
100
110
THRUST COEFFICIENT x 10⁴


DASHED LINE PRESENTS EXTRAPOLATED DATA
DENSITY ALTITUDE (FT)
CURVES DERIVED FROM FIGURE 6
12,800
11,120
10,000
8000
6000
5550
4000
BASE LINE 2160 FT
CURVE OBTAINED AT BASE LINE ALTITUDE FROM FIGURE 6
$C_p = A_0 + A_1 C_t^{1.5} + A_2 C_p$
$A_0 = 6.0891 \times 10^{-5}$
$A_1 = 1.0304$
ENGINE POWER COEFFICIENT $\times 10^4$
TORQUE COEFFICIENT $\times 10^4$

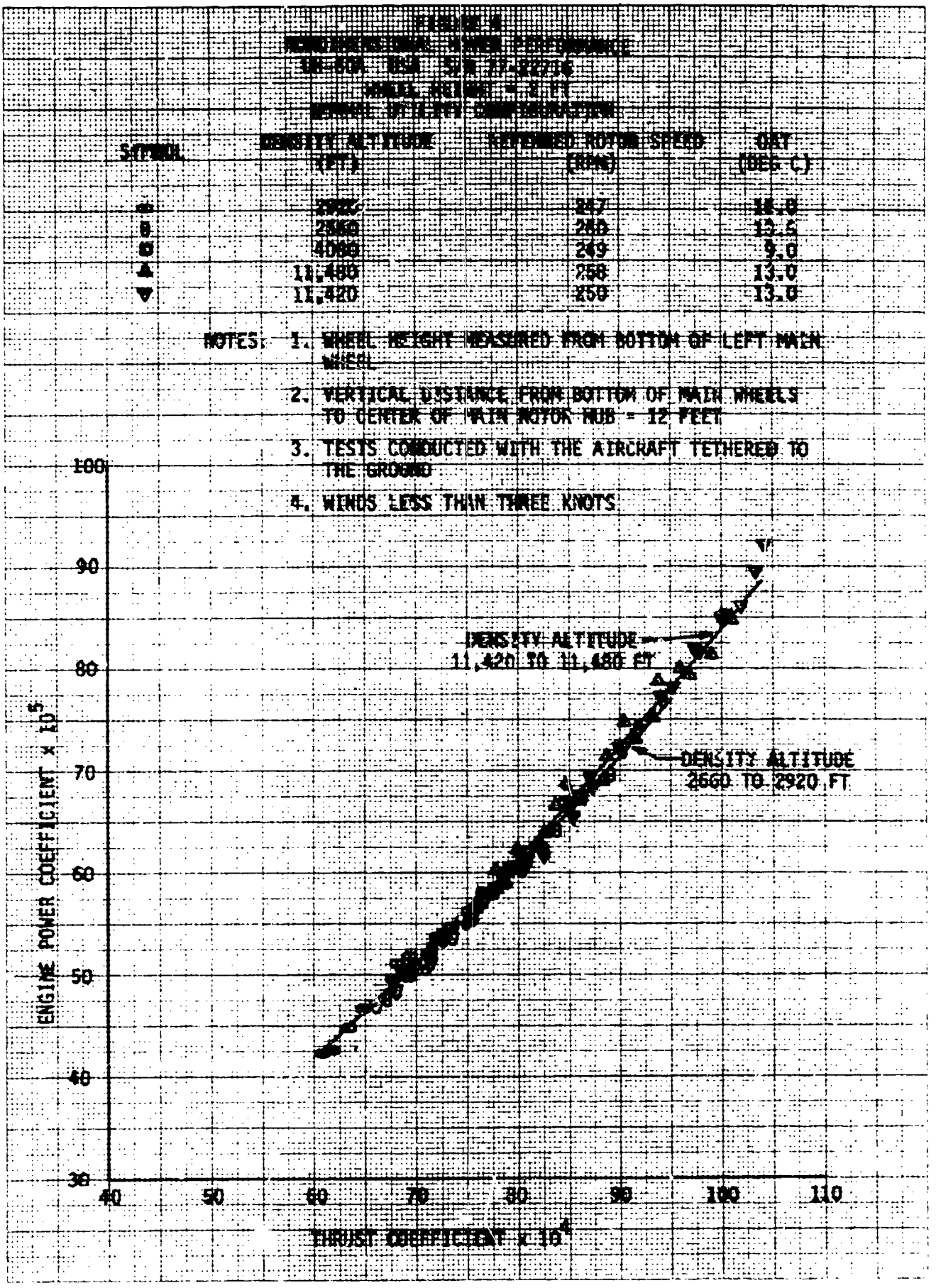
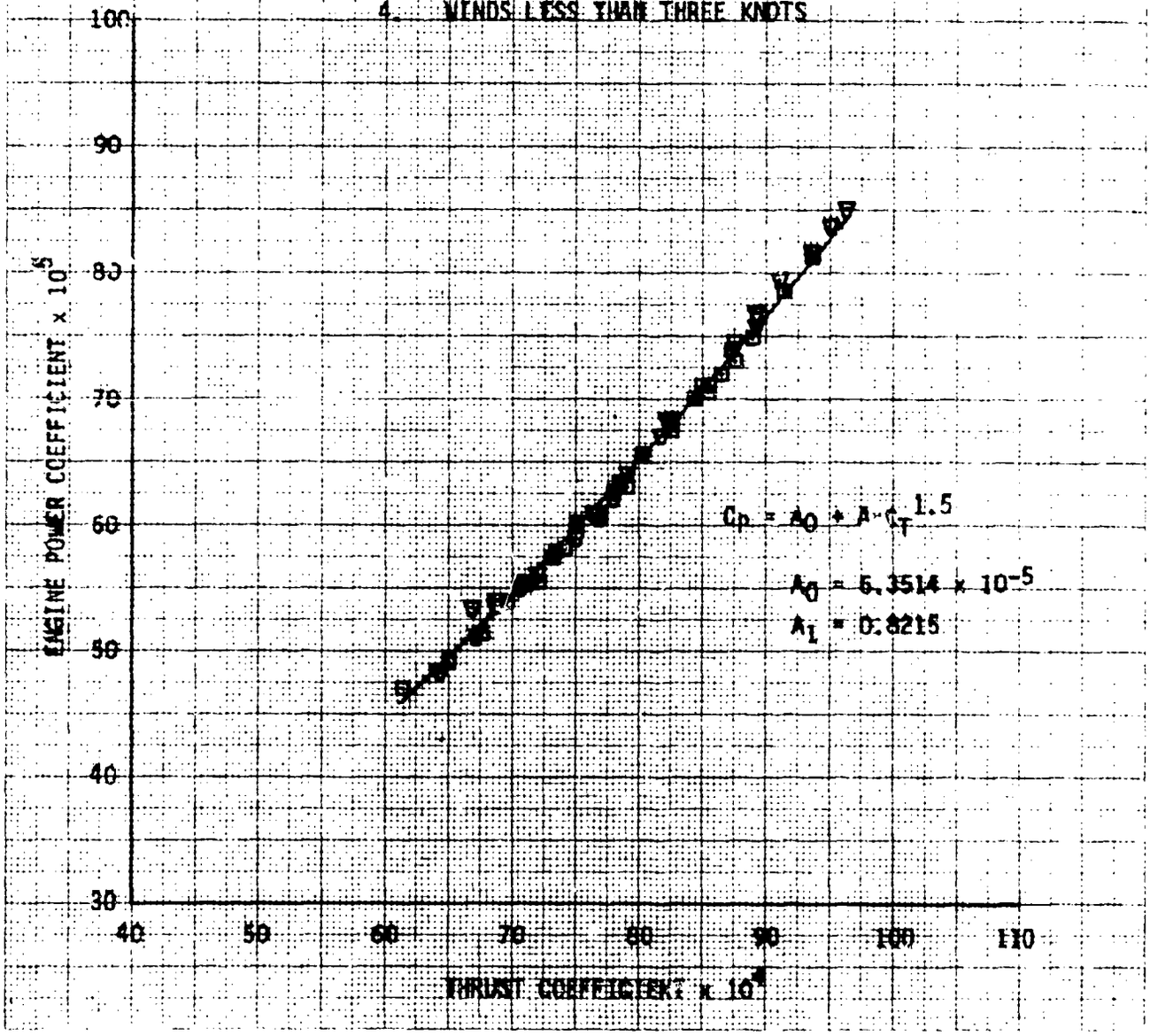

SYMBOL
DENSITY ALTITUDE (FT)
CORRECTED ROTOR SPEED (RPM)
OAT (DEG C)
247 15.0
2860 240 15.5
4060 249 9.0
11,480 258 13.0
11,420 250 13.0
NOTES:
1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. VERTICAL DISTANCE FROM BOTTOM OF MAIN WHEELS TO CENTER OF MAIN ROTOR HUB = 12 FEET
3. TESTS CONDUCTED WITH THE AIRCRAFT TETHERED TO THE GROUND
4. WINDS LESS THAN THREE KNOTS
DENSITY ALTITUDE 11,420 TO 11,480 FT
DENSITY ALTITUDE 2660 TO 2920 FT
ENGINE POWER COEFFICIENT x 10^5
THRUST COEFFICIENT x 10^4
100
90
80
70
60
50
40
30
40
50
60
70
80
90
100
110

FIGURE 5
REPRESENTATIVE ROTOR PERFORMANCE
UH-60A, USA S/N 77-22716
WHEEL HEIGHT = 5 FT
NORMAL UTILITY CONFIGURATION

SYMBOL	DENSITY ALTITUDE (FT)	CORRECTED ROTOR SPEED (RPM)	OAT (DEG C)
B	4740	257	23.0
A	4000	245	25.0

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 FEET
3. TESTS CONDUCTED WITH THE AIRCRAFT TETHERED TO THE GROUND
4. WINDS LESS THAN THREE KNOTS



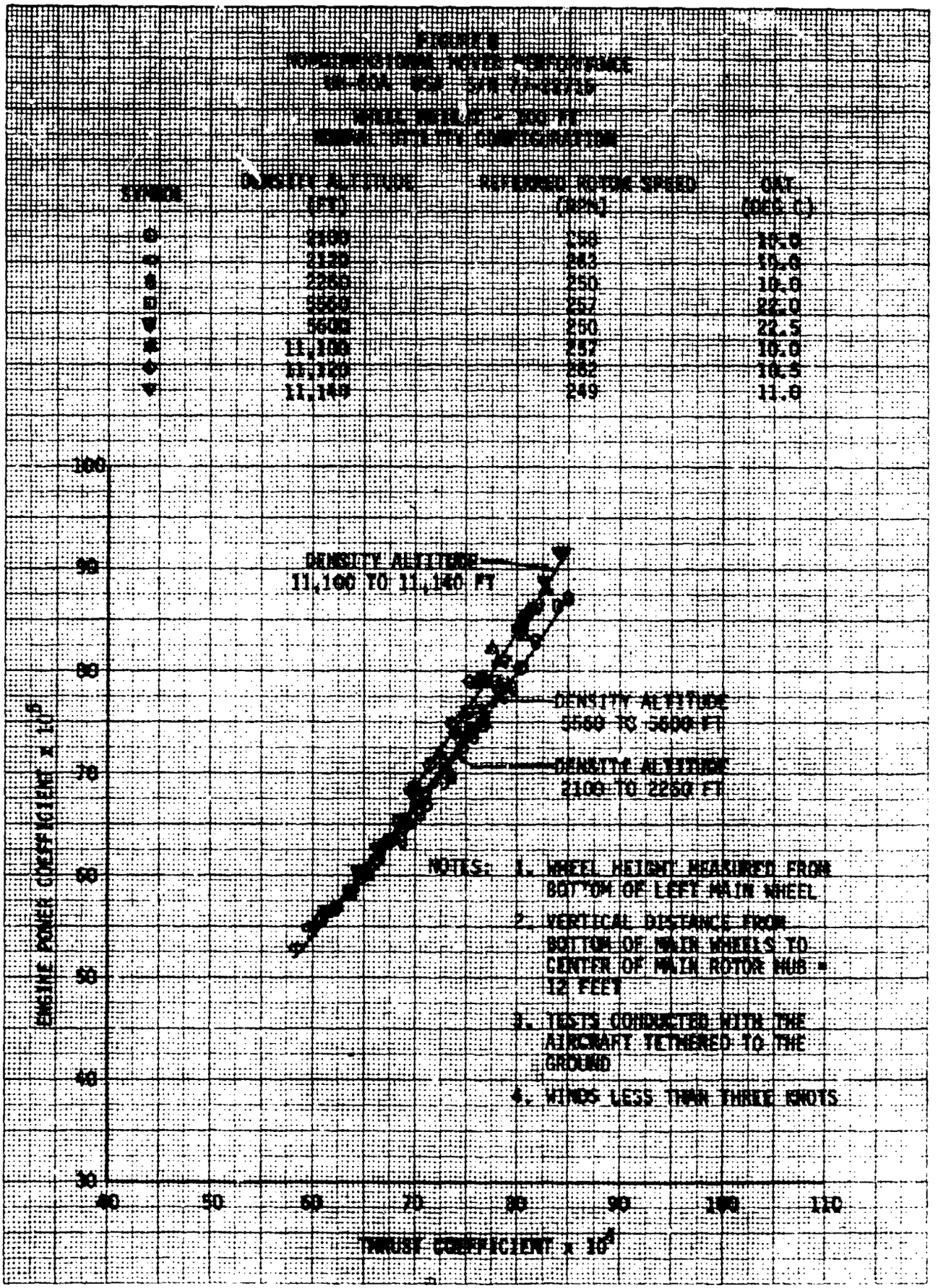

DENSITY ALTITUDE (FT)
ROTOR SPEED (RPM)
OAT (DEG C)
2100 250 10.0
2120 243 10.0
2250 250 10.0
3560 257 22.0
5600 250 22.5
11,100 257 10.0
11,140 262 10.5
11,149 249 11.0
DENSITY ALTITUDE 11,100 TO 11,140 FT
DENSITY ALTITUDE 3560 TO 5600 FT
DENSITY ALTITUDE 2100 TO 2250 FT
NOTES:
1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. VERTICAL DISTANCE FROM BOTTOM OF MAIN WHEELS TO CENTER OF MAIN ROTOR HUB = 12 FEET
3. TESTS CONDUCTED WITH THE AIRCRAFT TETHERED TO THE GROUND
4. WINDS LESS THAN THREE KNOTS
ENGINE POWER COEFFICIENT x 10^6
THRUST COEFFICIENT x 10^4
100
90
80
70
60
50
40
30
40
50
60
70
80
90
100
110

FIGURE 7

NONDIMENSIONAL ROTOR PERFORMANCE

UH-60A BSN 3FW 77-22718

WHEEL HEIGHT = 100 FT

CONFIGURATION: BR

SYMBOL	DENSITY ALTITUDE (FT)	REFERRED ROTOR SPEED (RPM)	OAT (DEG C)
⊖	3320	258	20.0
θ	3500	245	21.0

DASHED LINE DENOTES NORMAL UTILITY CONFIGURATION

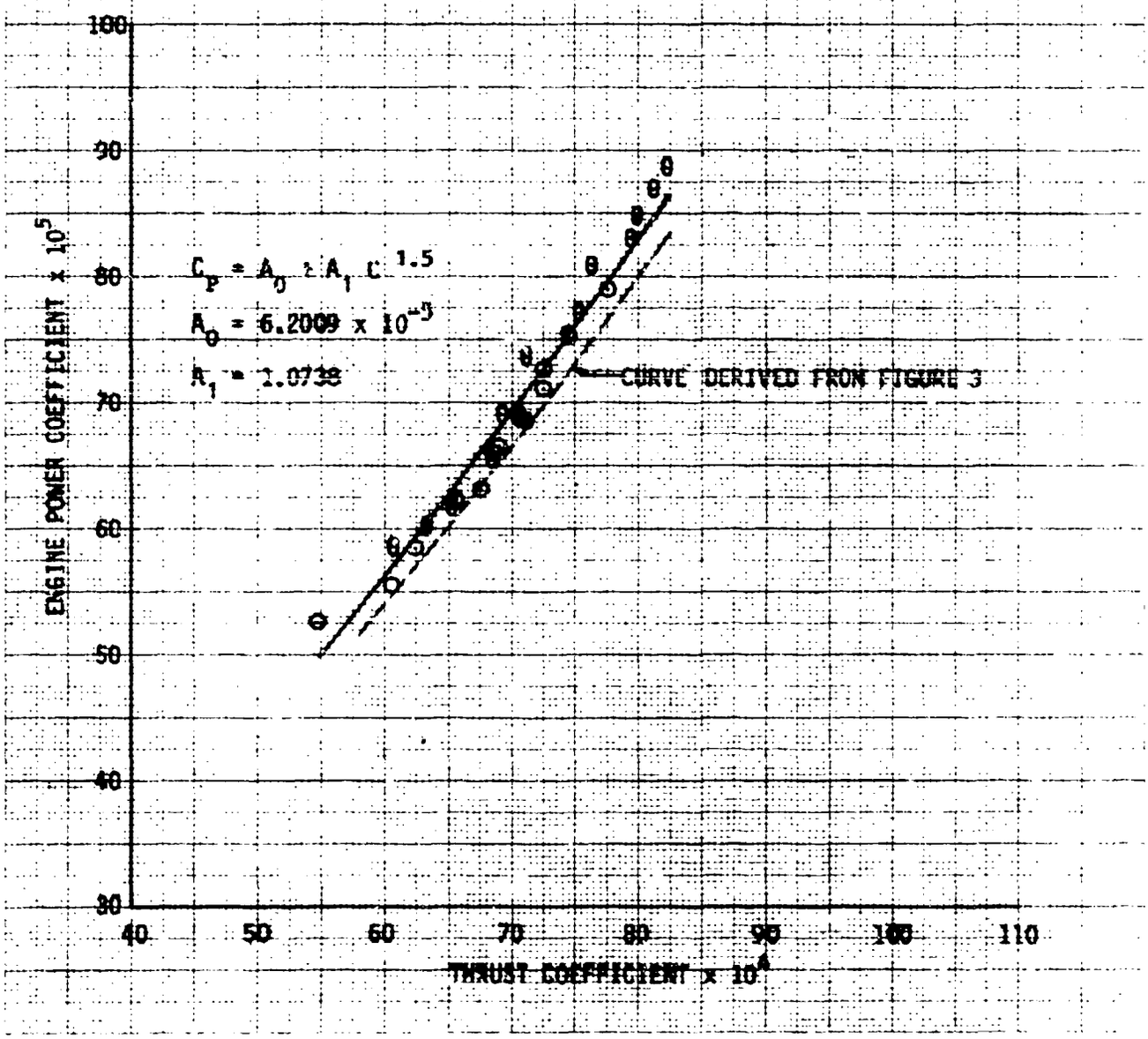
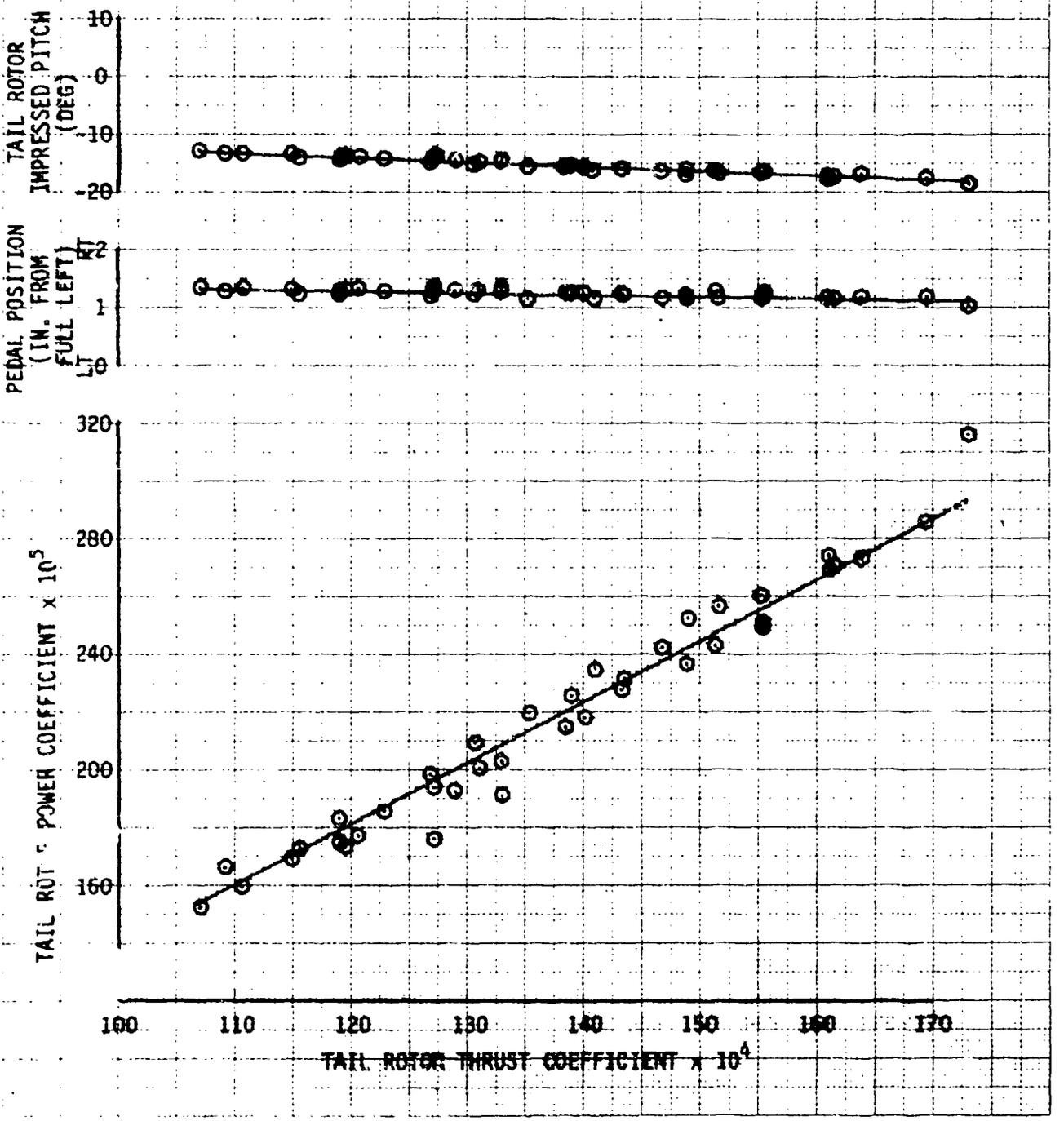


FIGURE 8
NONDIMENSIONAL TAIL ROTOR PERFORMANCE
UH-60A USA S/N 77-22716
WHEEL HEIGHT = 2 FT

SYMBOL	DENSITY ALTITUDE (FT)	REFERRED ROTOR SPEED (RPM)	OAT (DEG C)
⊙	11,480	258	13.0
⊙	15,420	250	13.0

NOTE: NORMAL UTILITY CONFIGURATION



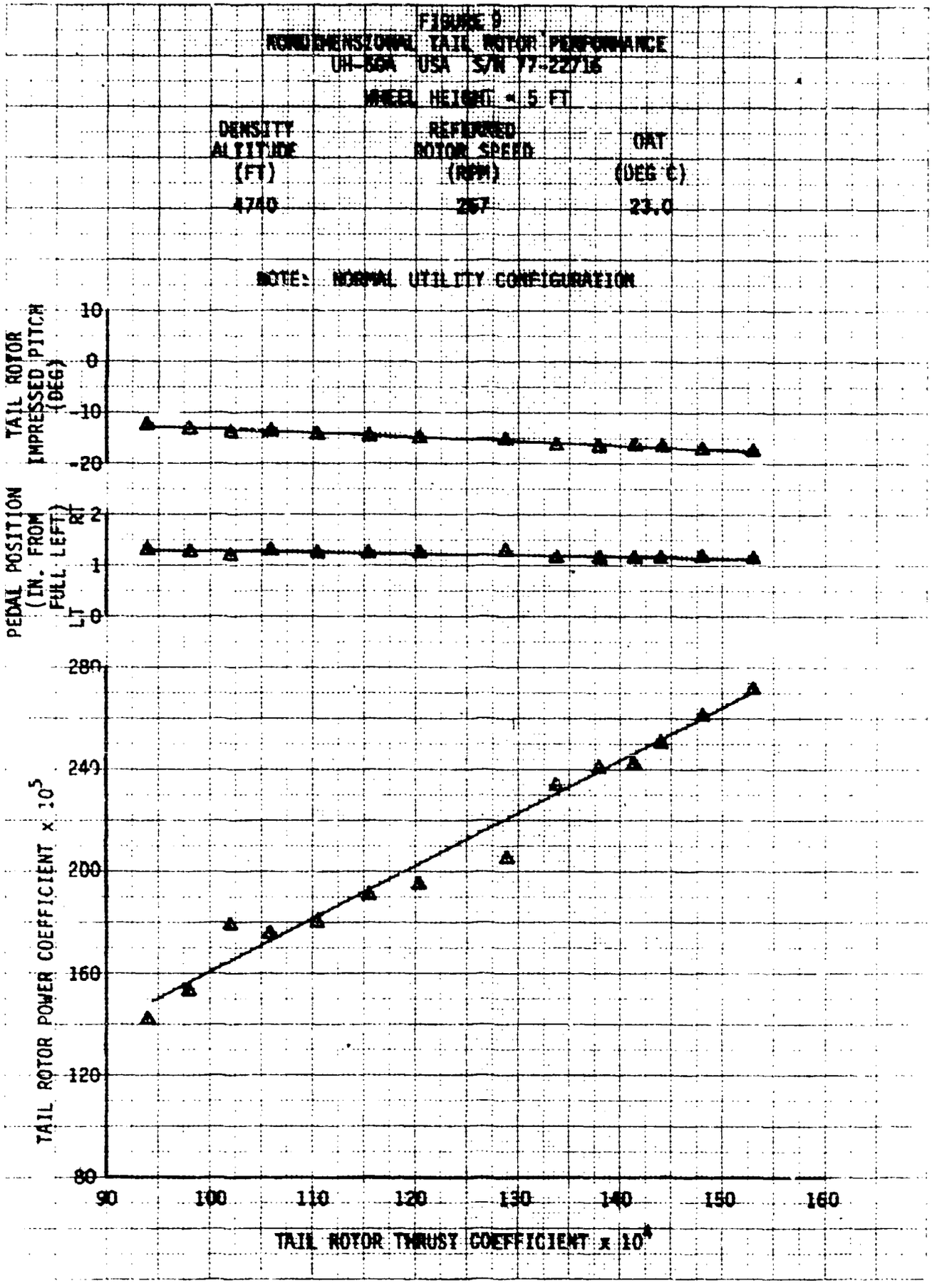


FIGURE 9
NONDIMENSIONAL TAIL ROTOR PERFORMANCE
UH-60A USA S/N 77-22716
WHEEL HEIGHT = 5 FT
DENSITY ALTITUDE (FT)
4740
REFERRED ROTOR SPEED (RPM)
257
OAT (DEG C)
23.0
NOTE: NORMAL UTILITY CONFIGURATION
TAIL ROTOR IMPRESSED PITCH (DEG)
10
0
-10
-20
PEDAL POSITION (IN. FROM FULL LEFT)
RT
LT
2
1
0
TAIL ROTOR POWER COEFFICIENT x 10^5
280
240
200
160
120
80
90
100
110
120
130
140
150
160
TAIL ROTOR THRUST COEFFICIENT x 10^4

FIGURE 10
NONDIMENSIONAL TAIL ROTOR PERFORMANCE
UH-60A USA S/N 77-22716

WHEEL HEIGHT = 100 FT

SYMBOL	DENSITY ALTITUDE (FT)	REFERRED ROTOR SPEED (RPM)	OAT (DEG C)
⊙	2100	258	10.0
⊙	2120	263	10.0
⊡	5560	257	22.0
△	5600	250	22.5
▽	11,100	257	10.0
◇	11,120	262	10.5
▽	11,140	249	11.0

NOTE: NORMAL UTILITY CONFIGURATION

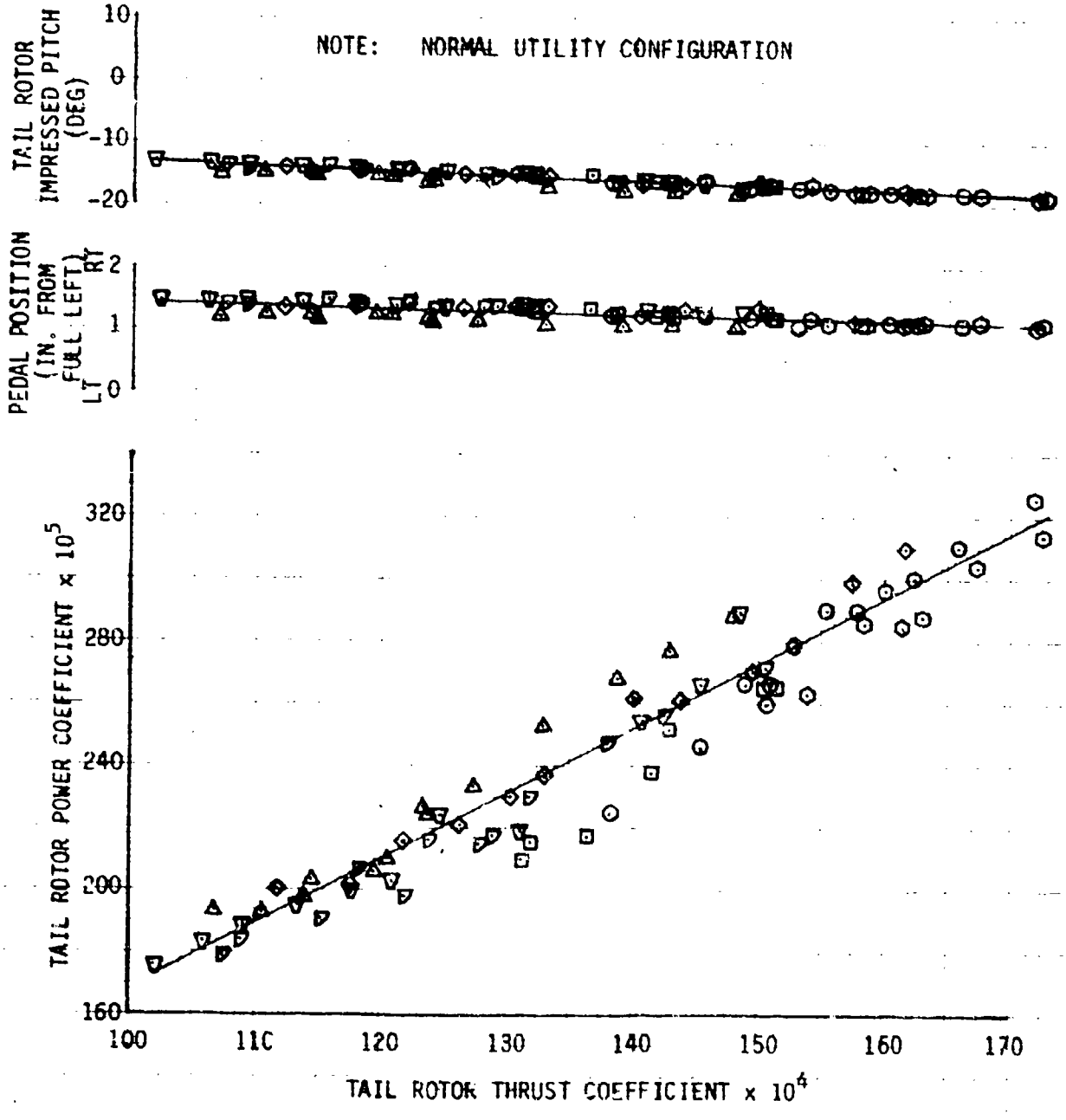


FIGURE 11
SUMMARY VERTICAL CLIMB PERFORMANCE
UH-60A USA S/N 77-22716
NORMAL UTILITY CONFIGURATION
95 PERCENT INTERMEDIATE RATED POWER
GROSS WEIGHT = 16,260 LB
AMBIENT TEMPERATURE = 35° C

NOTE: DATA DERIVED FROM FIGURE 12, APP E, AND FIGURE 2, APP B

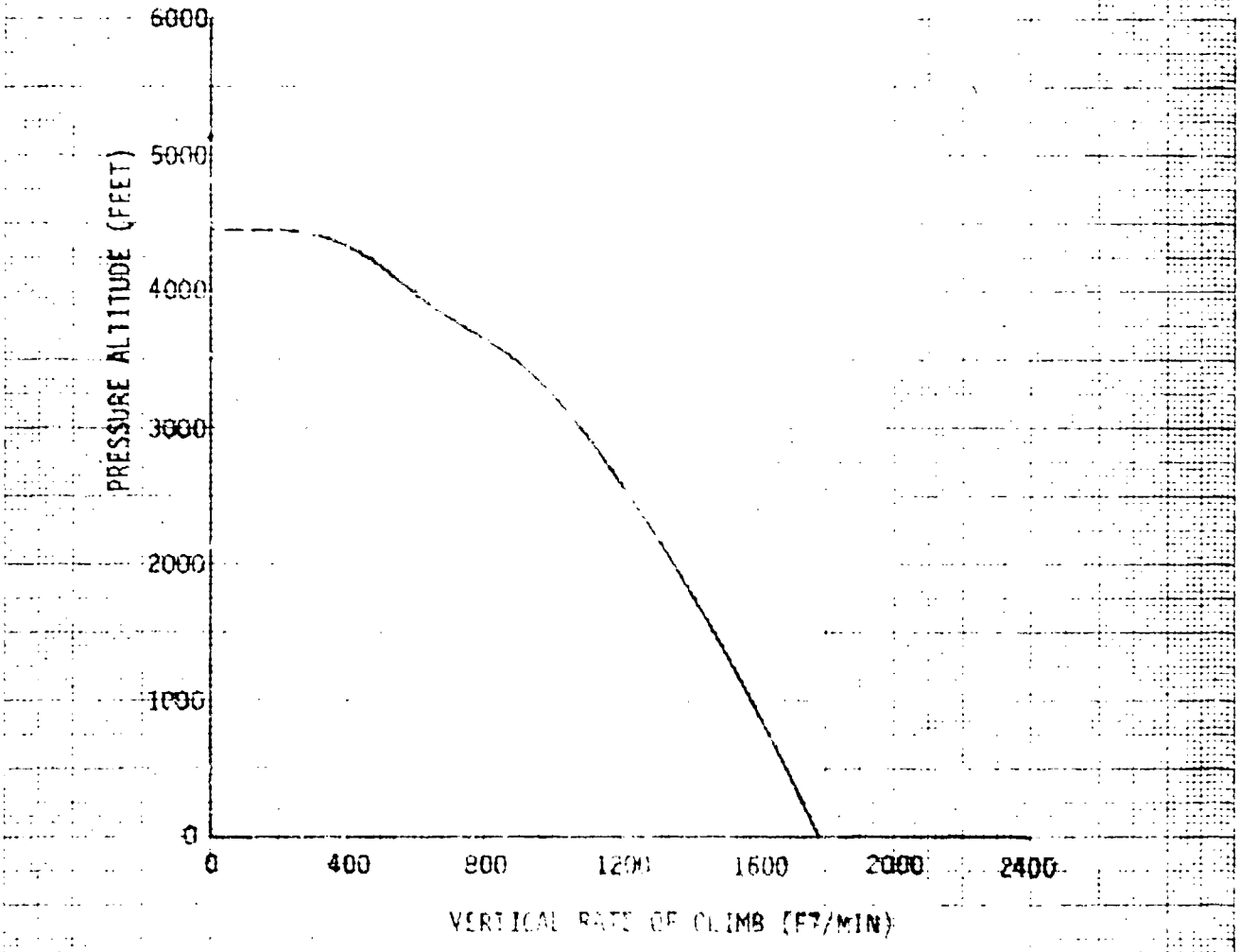


FIGURE B2
NONDIMENSIONAL VERTICAL CLIMB PERFORMANCE
UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T
◇	18,380	347.0(FWD)	0.2(RT)	2540	12.0	253	0.007656
□	17,380	347.5(FWD)	0.0(MID)	3180	17.0	254	0.006972
△	16,000	347.0(FWD)	0.0(MID)	4700	24.0	256	0.006615

NOTE: NORMAL UTILITY CONFIGURATION

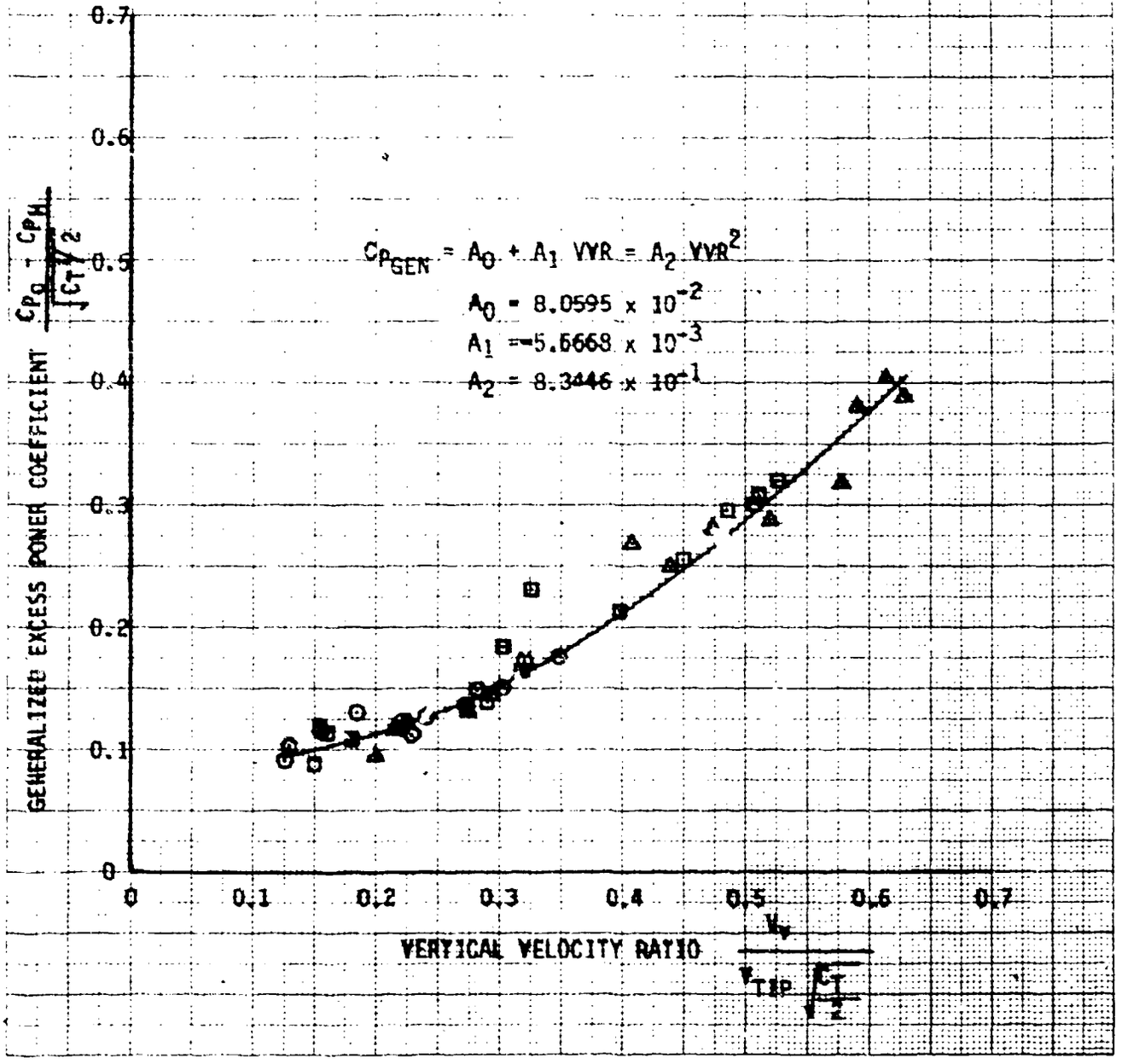


FIGURE 13
VARIATION IN RATE OF CLIMB AS A FUNCTION OF SHAFT HORSEPOWER
UH-60A USA S/N 76-22716

SYMBOL	DENSITY ALTITUDE (FT)	GROSS WEIGHT (LB)	ROTOR SPEED (RPM)
○	2500	16,500	258
□	5000	16,500	258
△	7500	16,500	258
◇	10,000	16,500	259
▽	12,500	16,500	258
▽	15,000	16,500	258

NOTES: 1. TESTS CONDUCTED AT BEST RATE OF CLIMB SPEED AS DETERMINED FROM FIGURES 34 THROUGH 36

2. $K_p = \frac{\Delta R/C}{\Delta SHP} \times \frac{GW}{33000} = 0.76$

3. NORMAL UTILITY CONFIGURATION

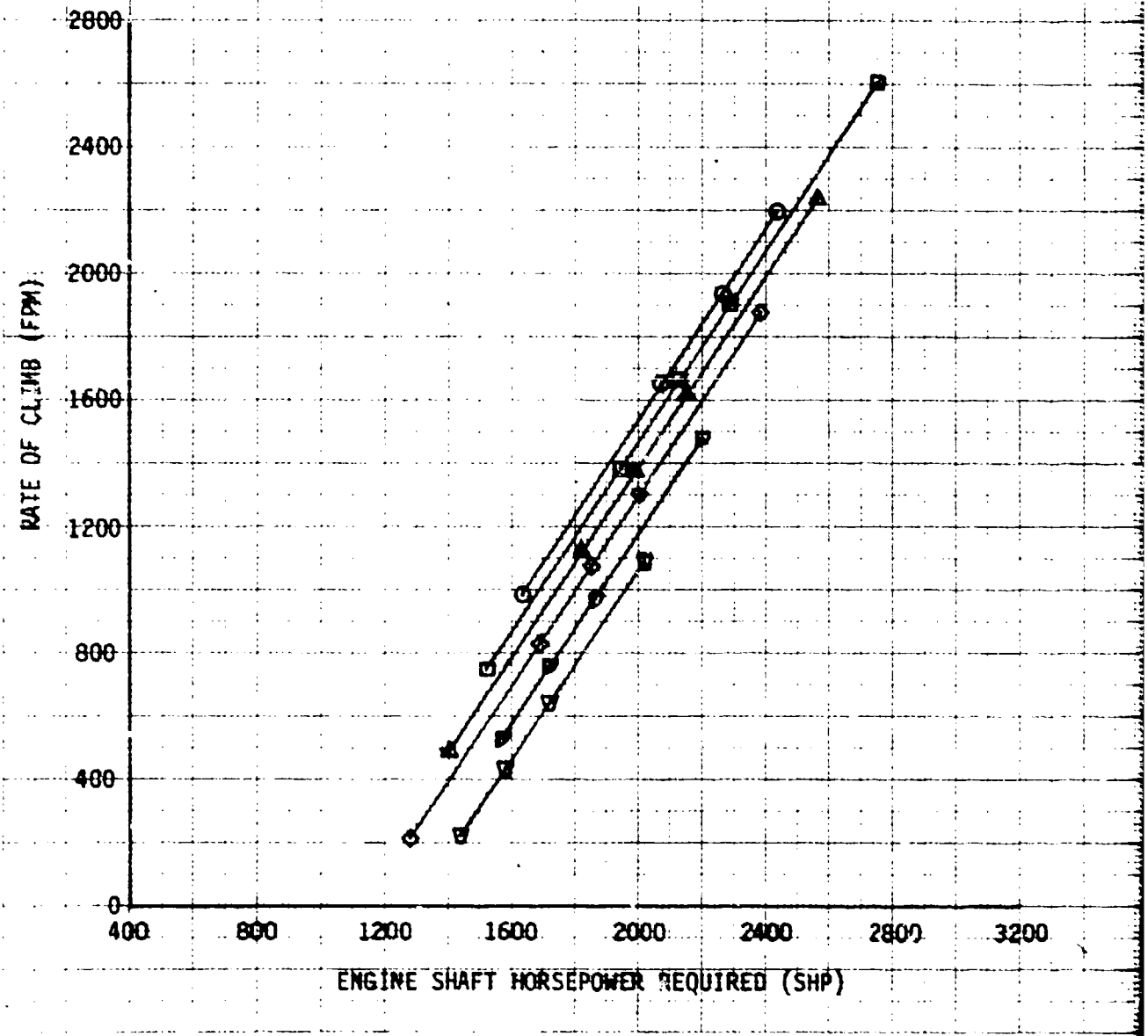
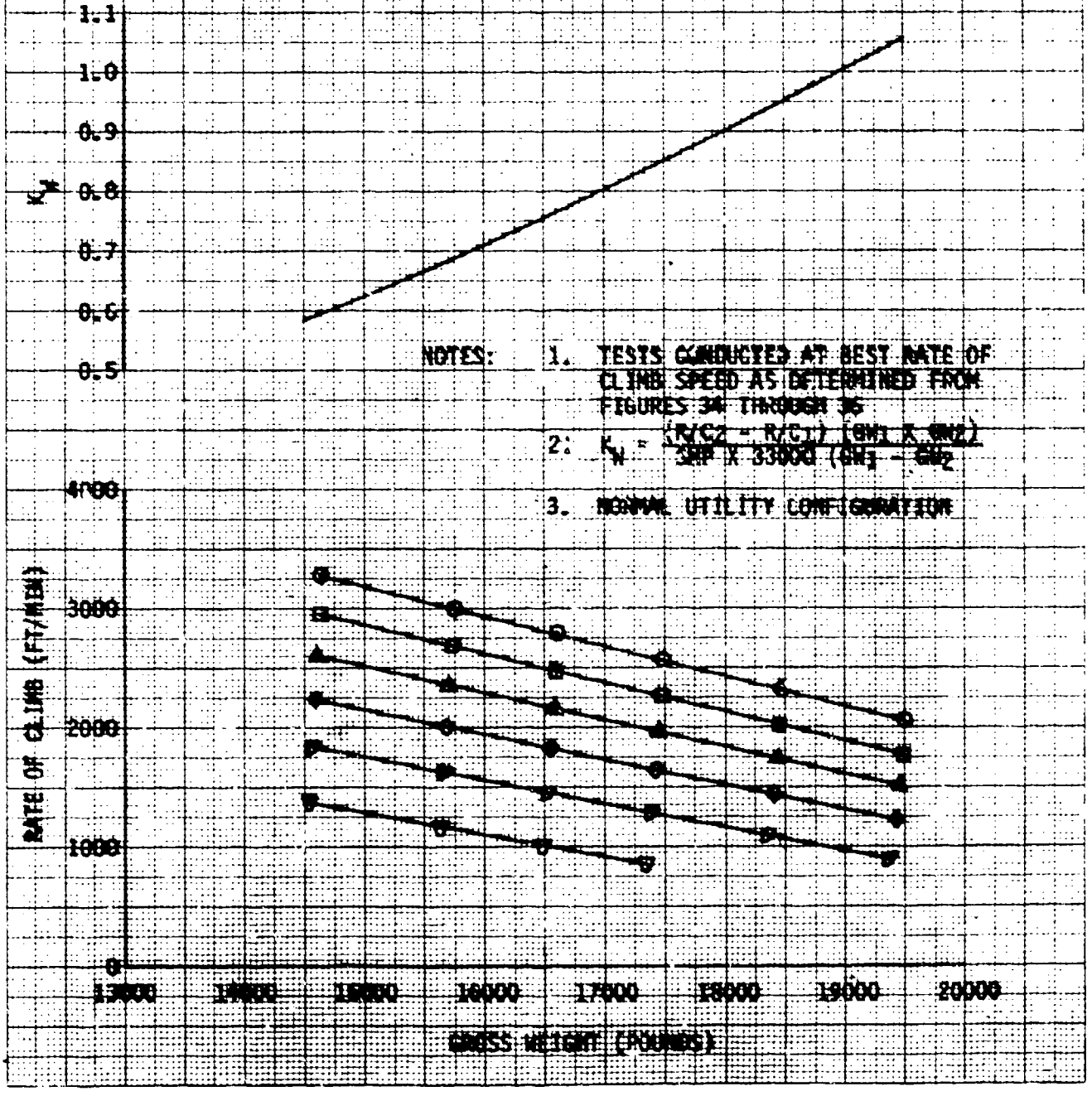
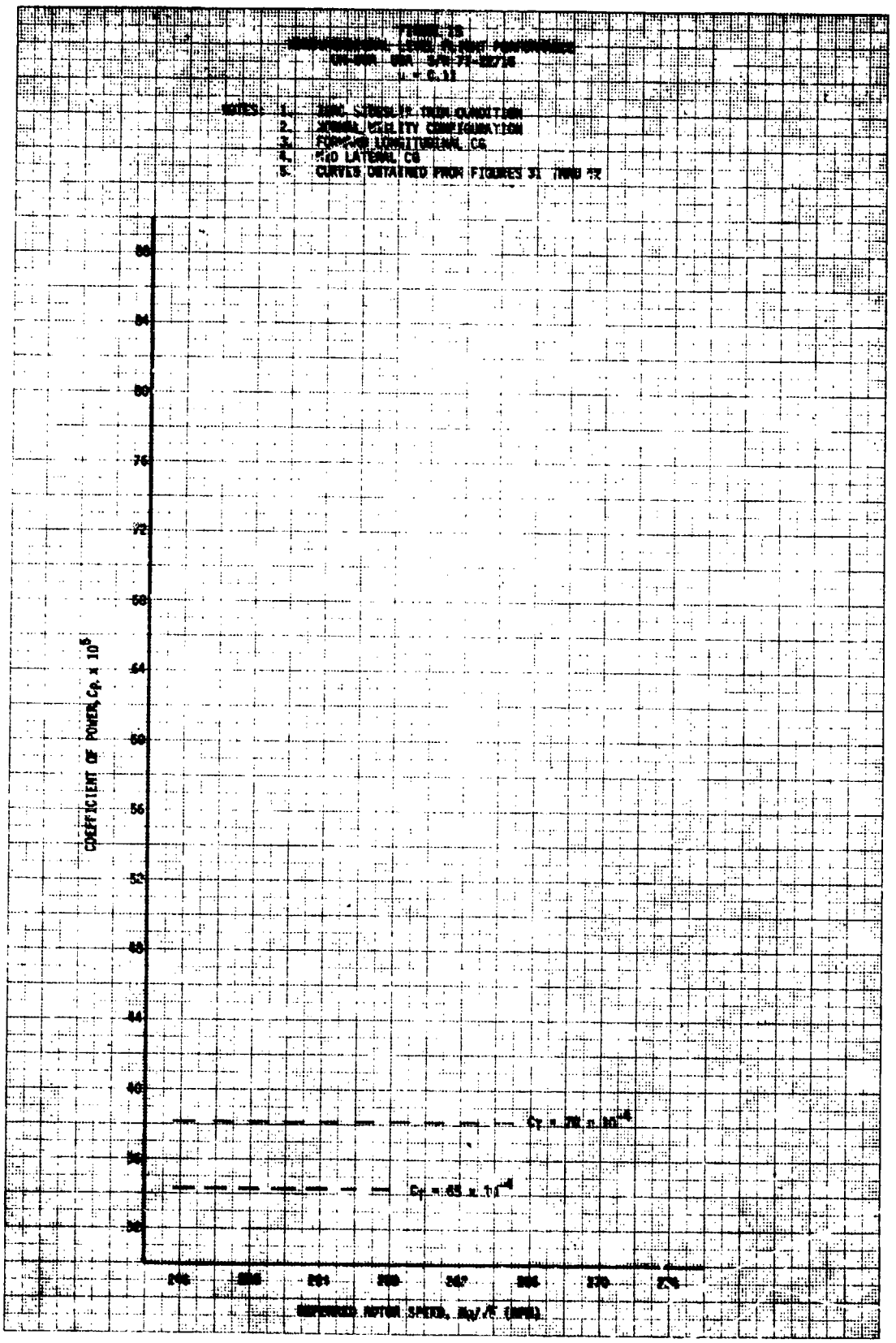


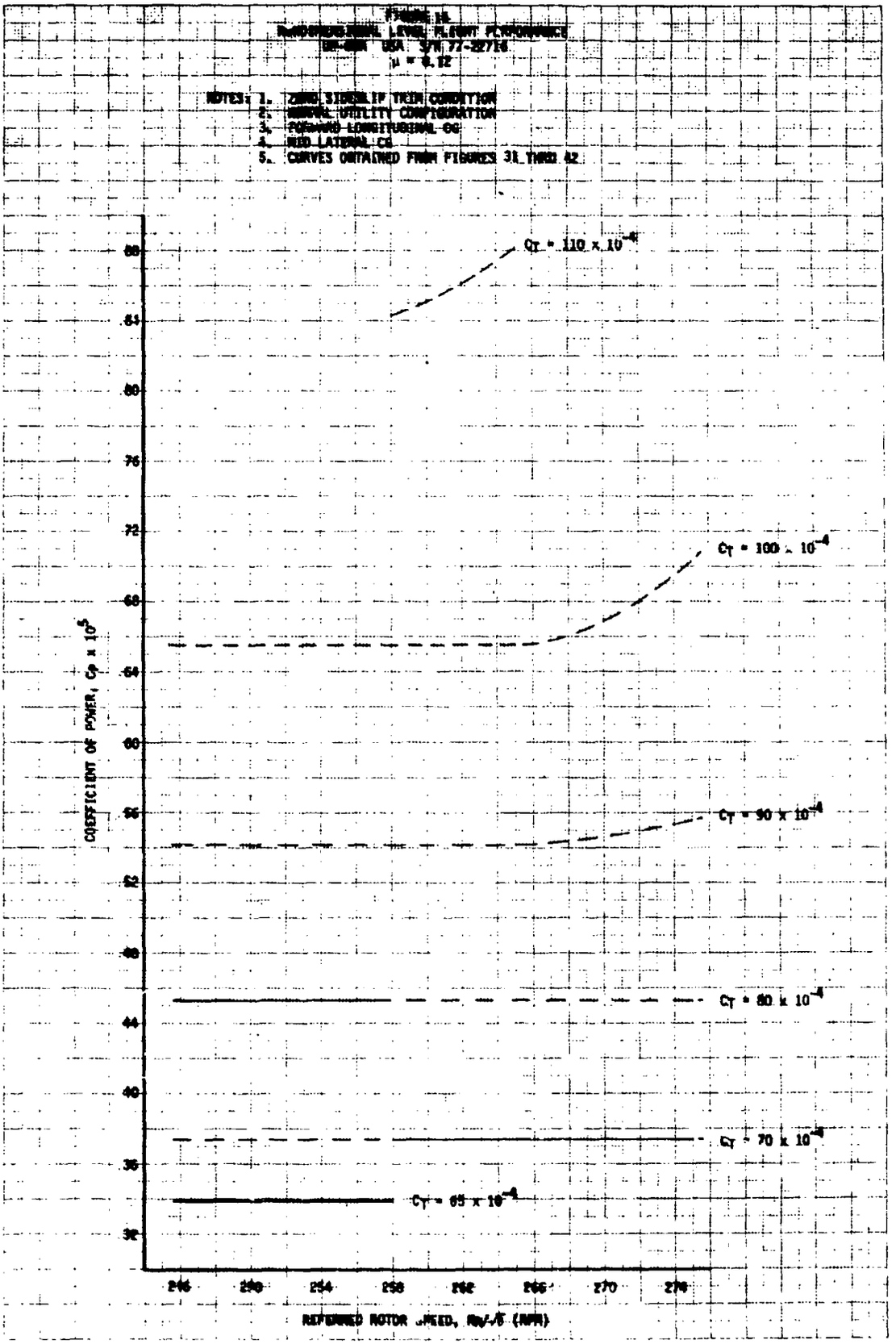
FIGURE 34
VARIATION IN RATE OF CLIMB AS A FUNCTION OF GROSS WEIGHT
UH-60A USA S/N 76-22716

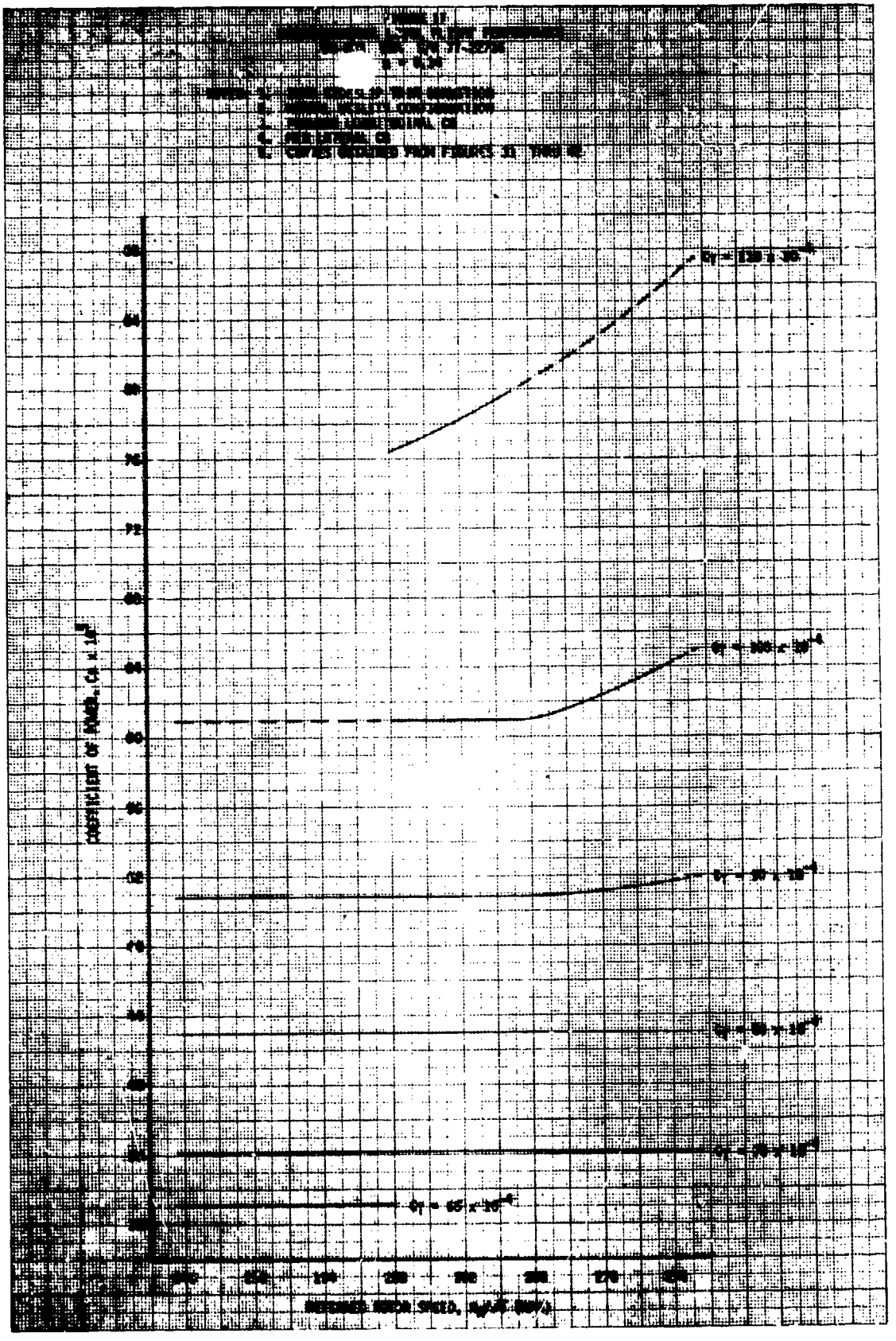
SYMBOL	DENSITY ALTITUDE (FT)	ROTOR SPEED (RPM)	SHAFT HORSEPOWER (SHP)
○	2500	258	2075
□	5000	258	2097
△	7500	258	2519
◇	10,000	258	2342
▽	12,500	258	2164
▽	15,000	258	1906



NOTES:
1. TESTS CONDUCTED AT BEST RATE OF CLIMB SPEED AS DETERMINED FROM FIGURES 34 THROUGH 36
2. $K_W = \frac{(R/C2 - R/C1)\ (GW1 \times GW2)}{SHP \times 33000\ (GW_1 - GW_2}$
3. NORMAL UTILITY CONFIGURATION


NOTES:
1.
2.
3.
4.
5.
COEFFICIENT OF POWER, $C_p \times 10^5$
$C_T = 70 \times 10^{-6}$
$C_T = 65 \times 10^{-6}$


NOTES: 1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. CURVES OBTAINED FROM FIGURES 31 THRU 42
$C_T = 110 \times 10^{-4}$
$C_T = 100 \times 10^{-4}$
$C_T = 90 \times 10^{-4}$
$C_T = 80 \times 10^{-4}$
$C_T = 70 \times 10^{-4}$
$C_T = 65 \times 10^{-4}$
COEFFICIENT OF POWER, $C_P \times 10^5$
REFERRED ROTOR SPEED, $\Omega_R/\sqrt{\theta}$ (RPM)



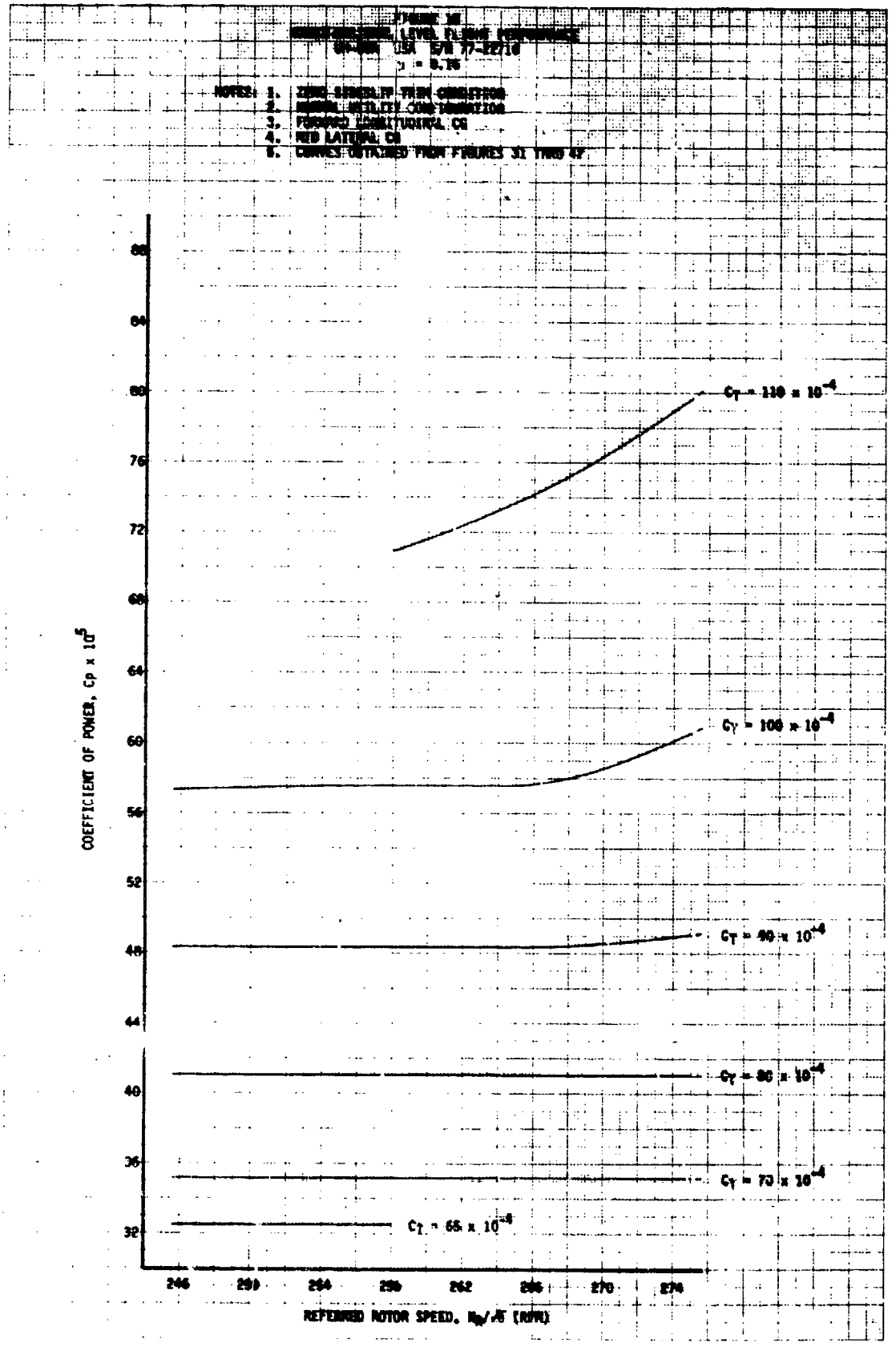

NOTES: 1.
2.
3. FORWARD HORIZONTAL CG
4. AFD LATERAL CG
5. CURVES OBTAINED FROM FIGURES 31 THRU 47
C_T = 110 x 10^{-6}
C_T = 100 x 10^{-6}
C_T = 90 x 10^{-6}
C_T = 80 x 10^{-6}
C_T = 70 x 10^{-6}
C_T = 65 x 10^{-6}
COEFFICIENT OF POWER, C_P x 10^5
REFERRED ROTOR SPEED, $\Omega_R/\sqrt{\theta}$ (RPM)

FIGURE 19
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716
$\mu = 0.18$

NOTES: 1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. CURVES OBTAINED FROM FIGURES 31 THRU 42

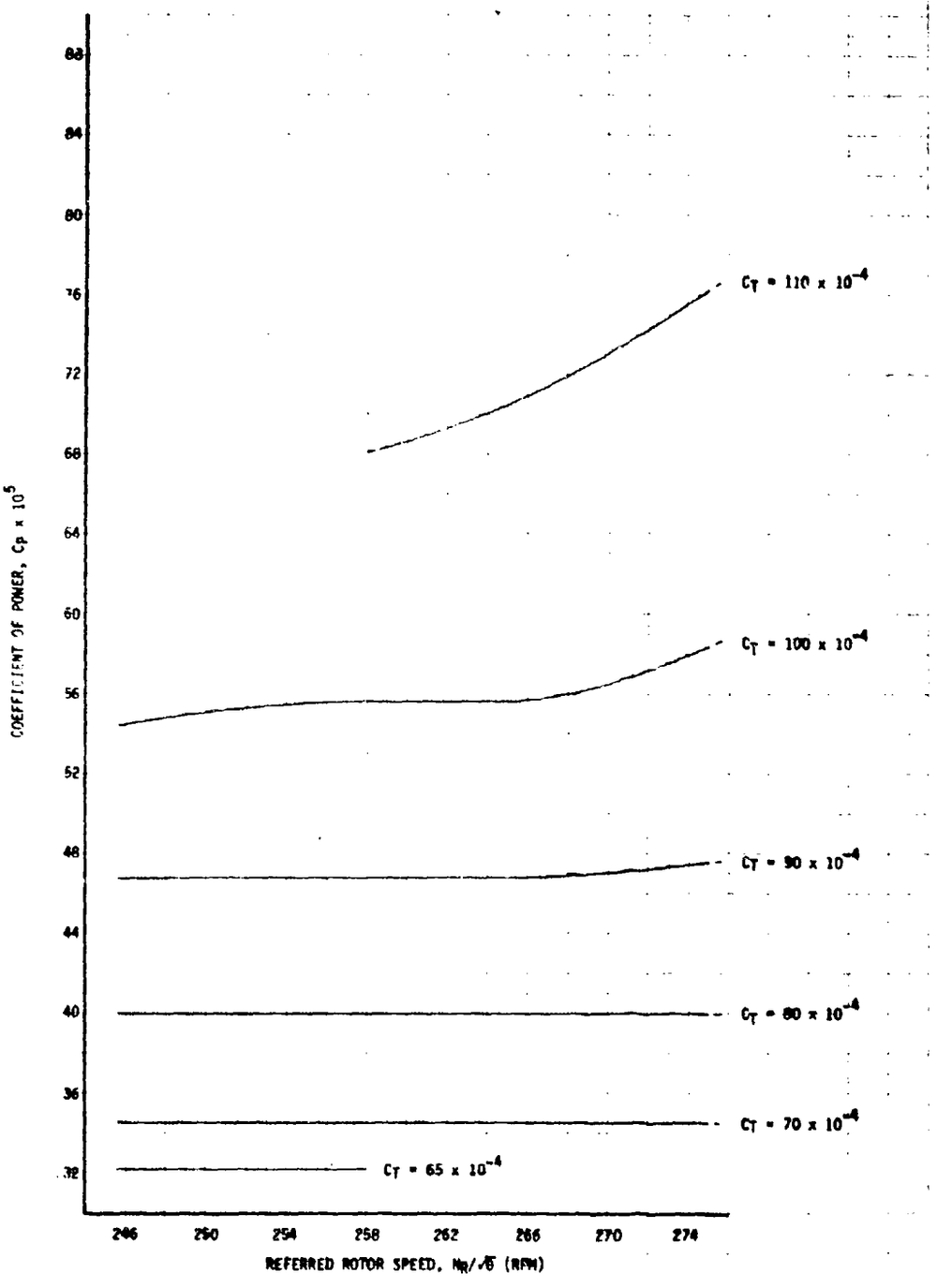
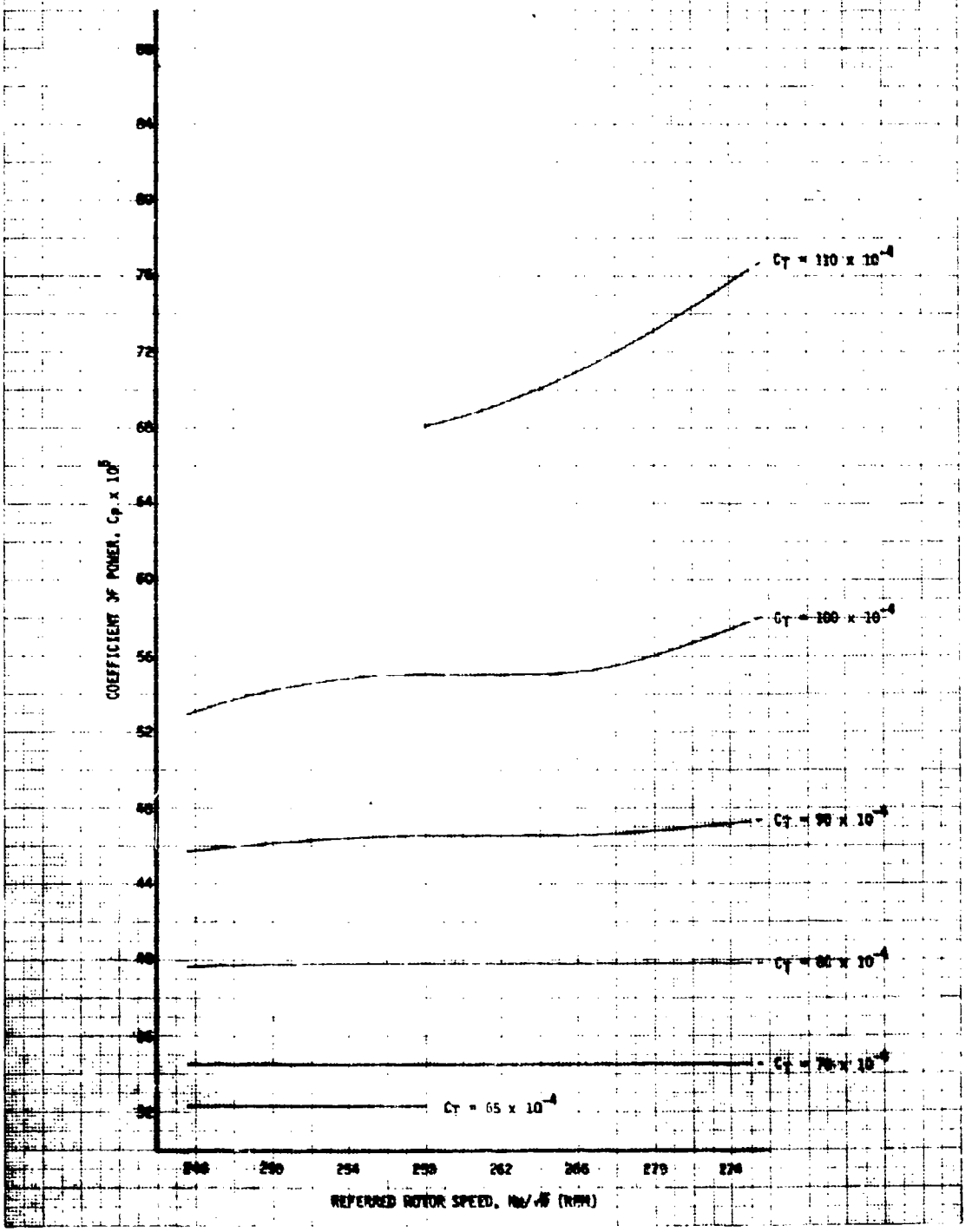


FIGURE 20
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A [illegible] S/N 77-22716
μ = 0.30

NOTES: 1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. CURVES OBTAINED FROM FIGURES 31 THRU 42



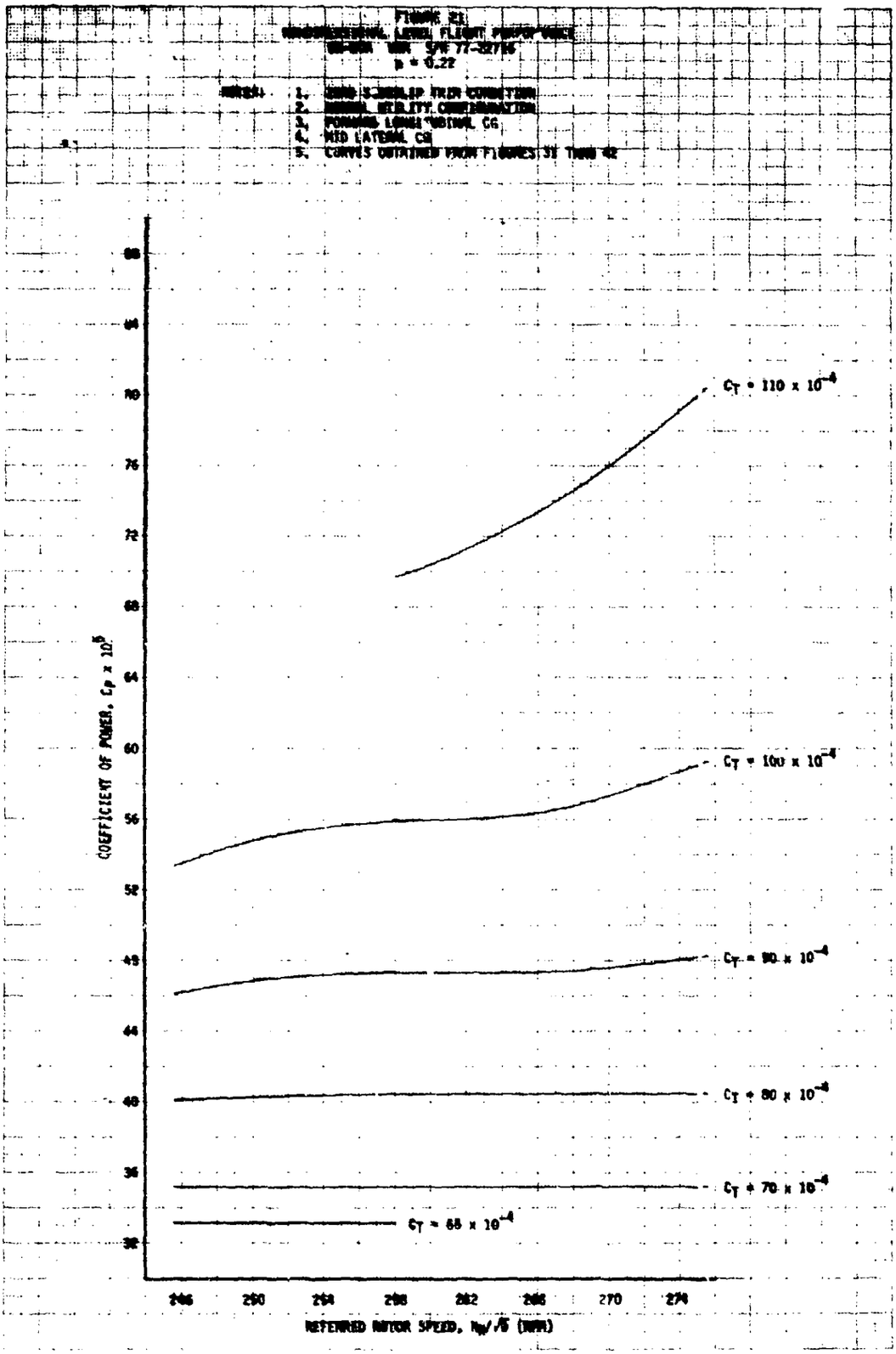
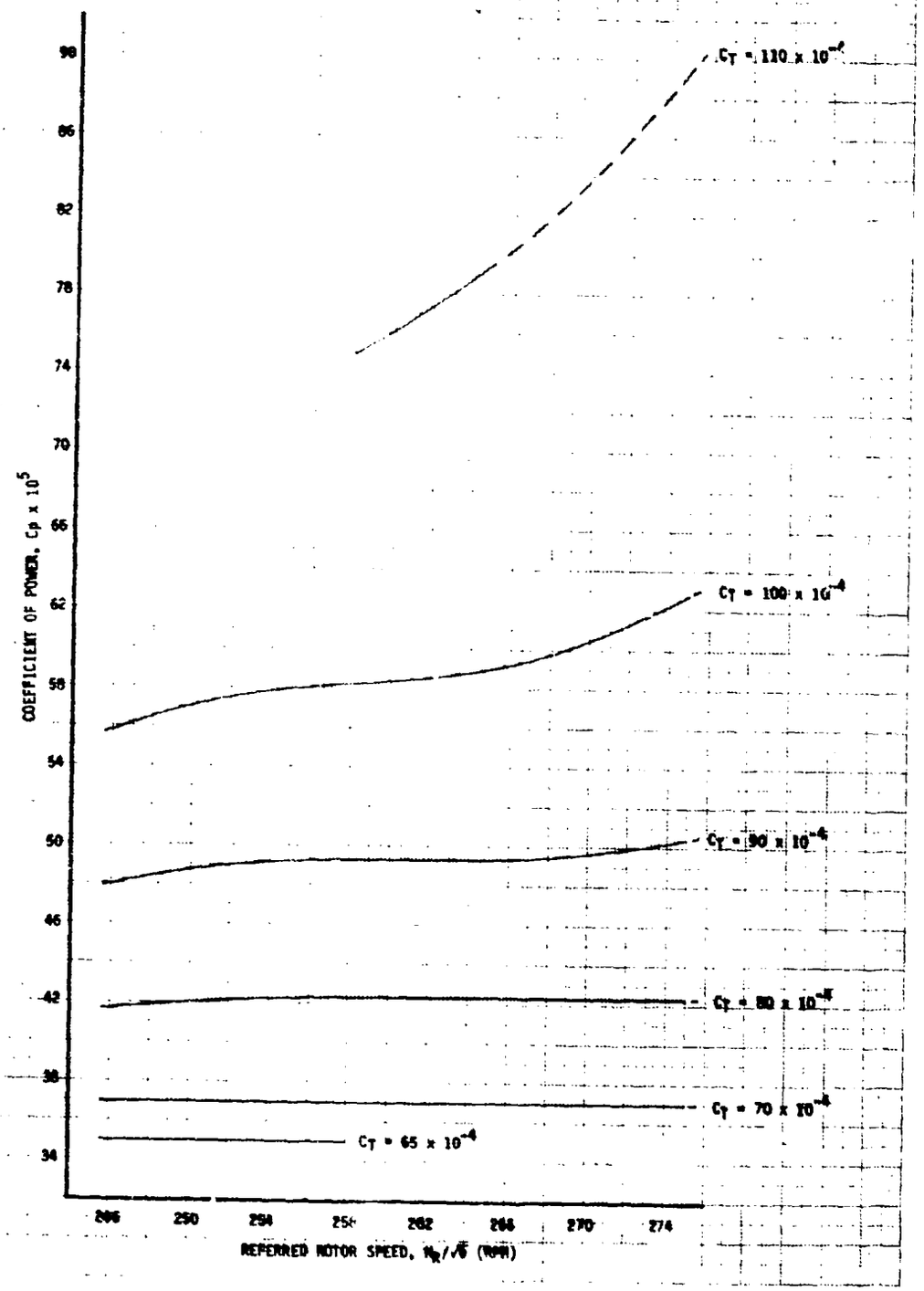
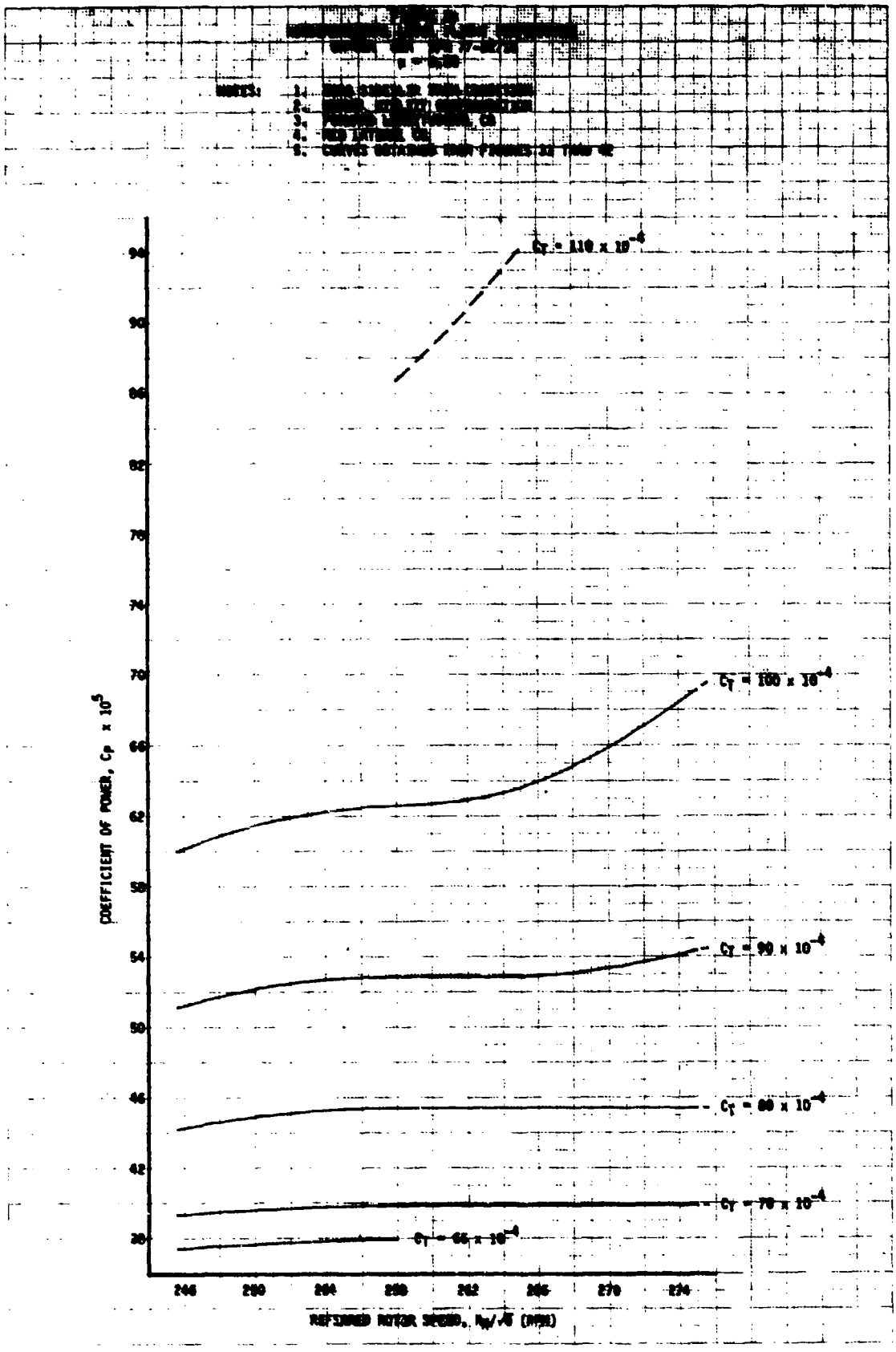

C_T = 110 x 10^{-4}
C_T = 100 x 10^{-4}
C_T = 90 x 10^{-4}
C_T = 80 x 10^{-4}
C_T = 70 x 10^{-4}
C_T = 60 x 10^{-4}
COEFFICIENT OF POWER, C_P x 10^5
RETENRED ROTOR SPEED, $\Omega_R/\sqrt{\theta}$ (RPM)

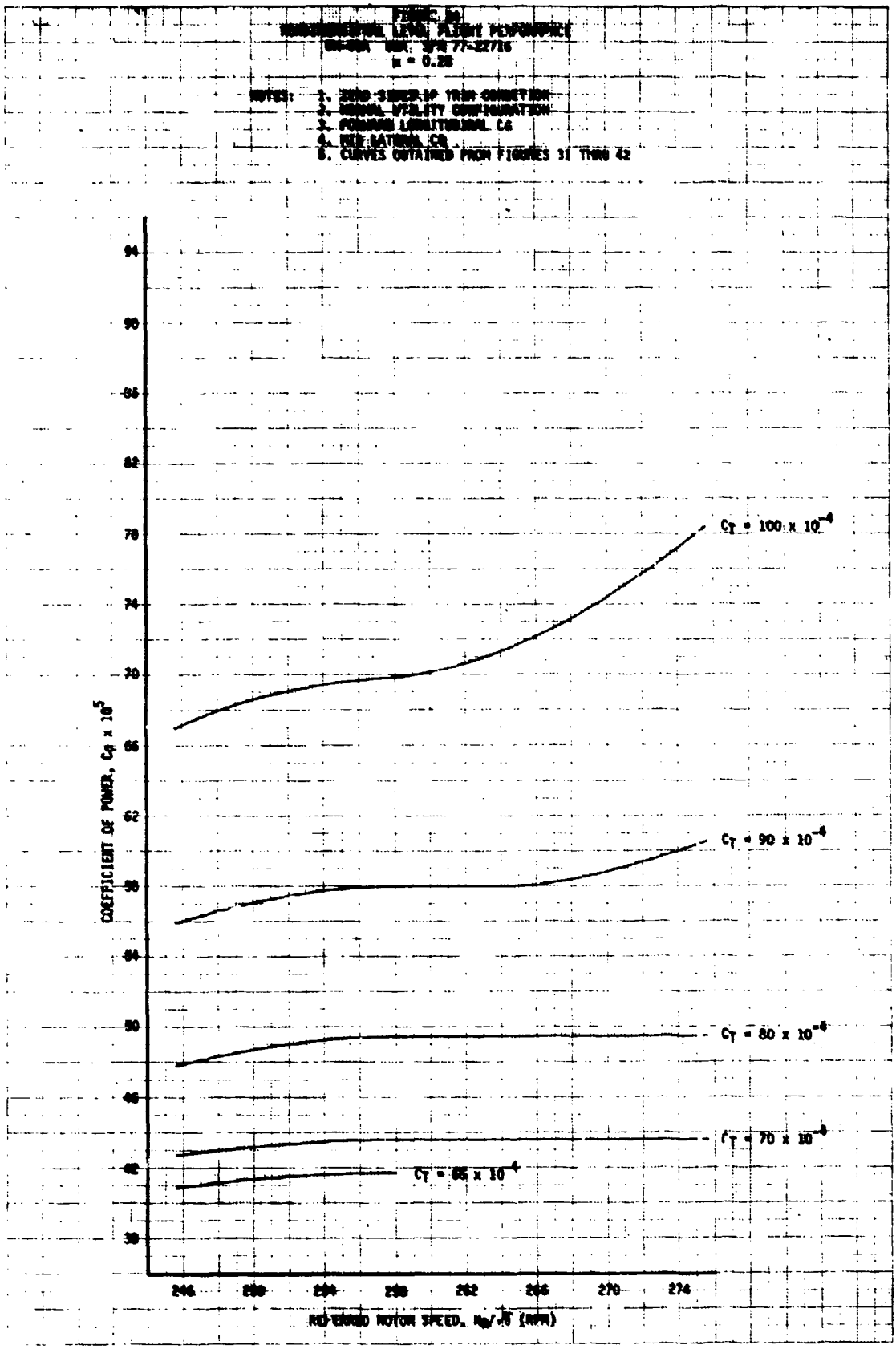
FIGURE 22

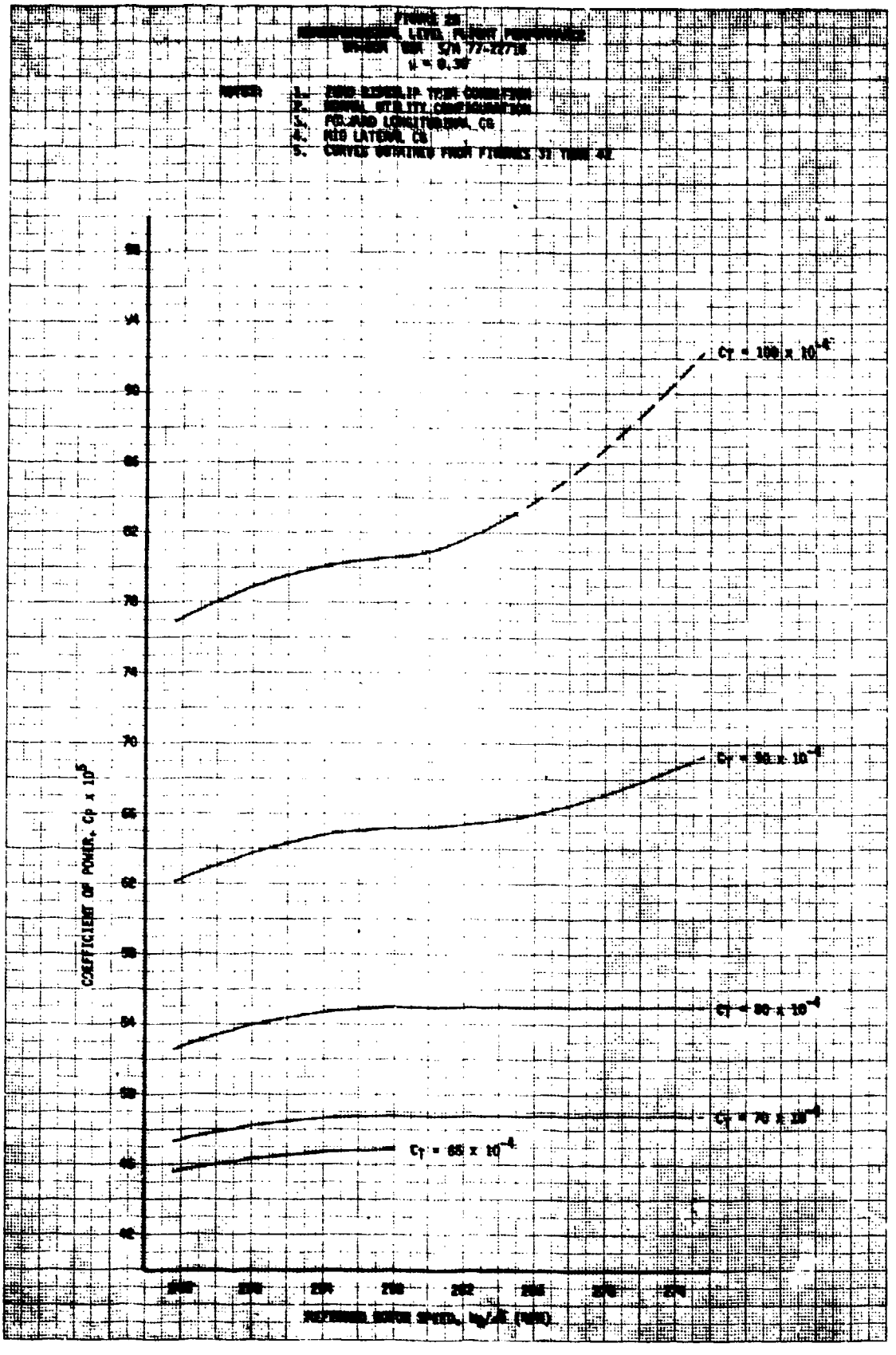
AERODYNAMICAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716
μ = 0.25

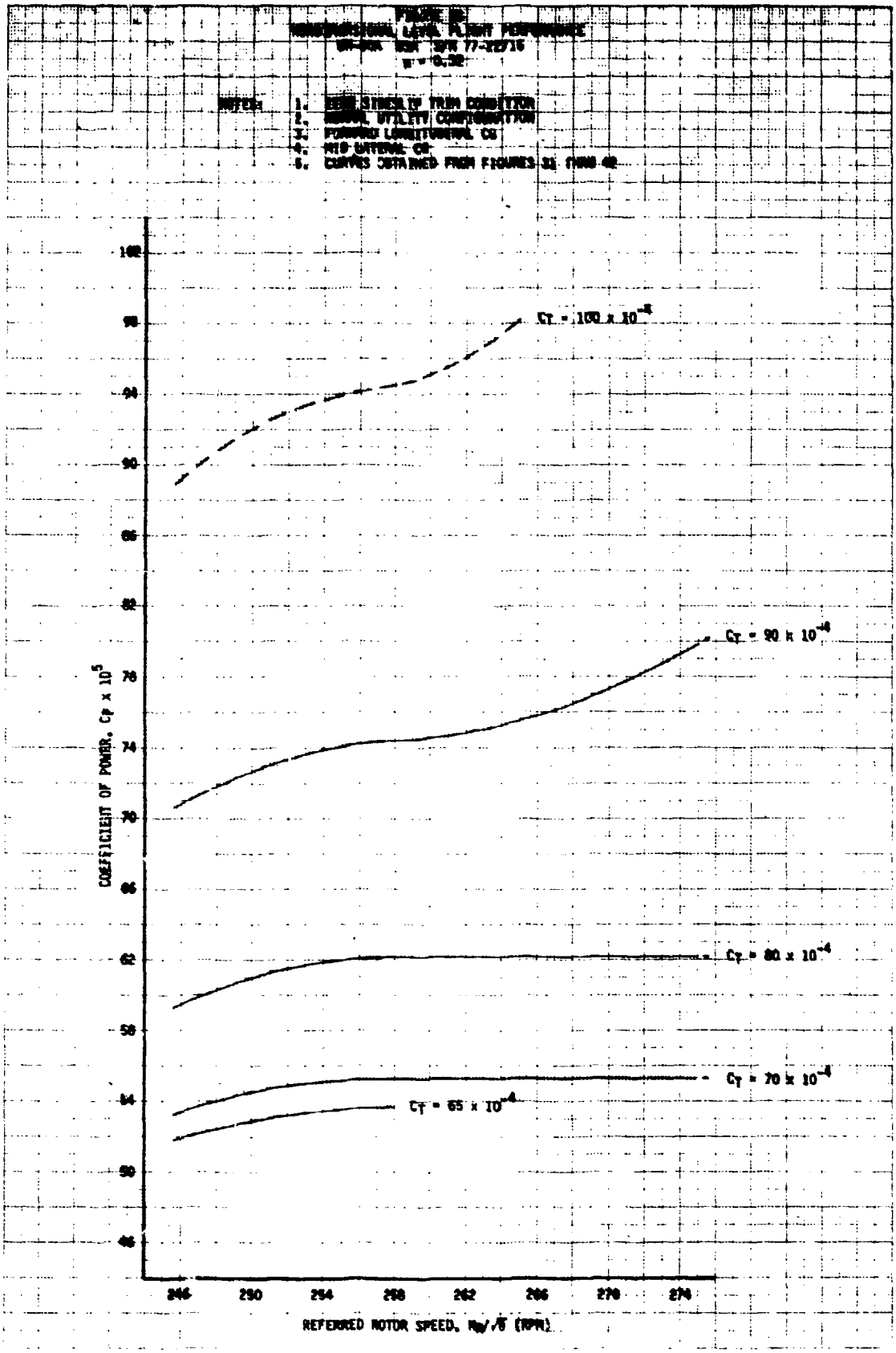
NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NOMINAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. CURVES OBTAINED FROM FIGURES 31 THRU 42.

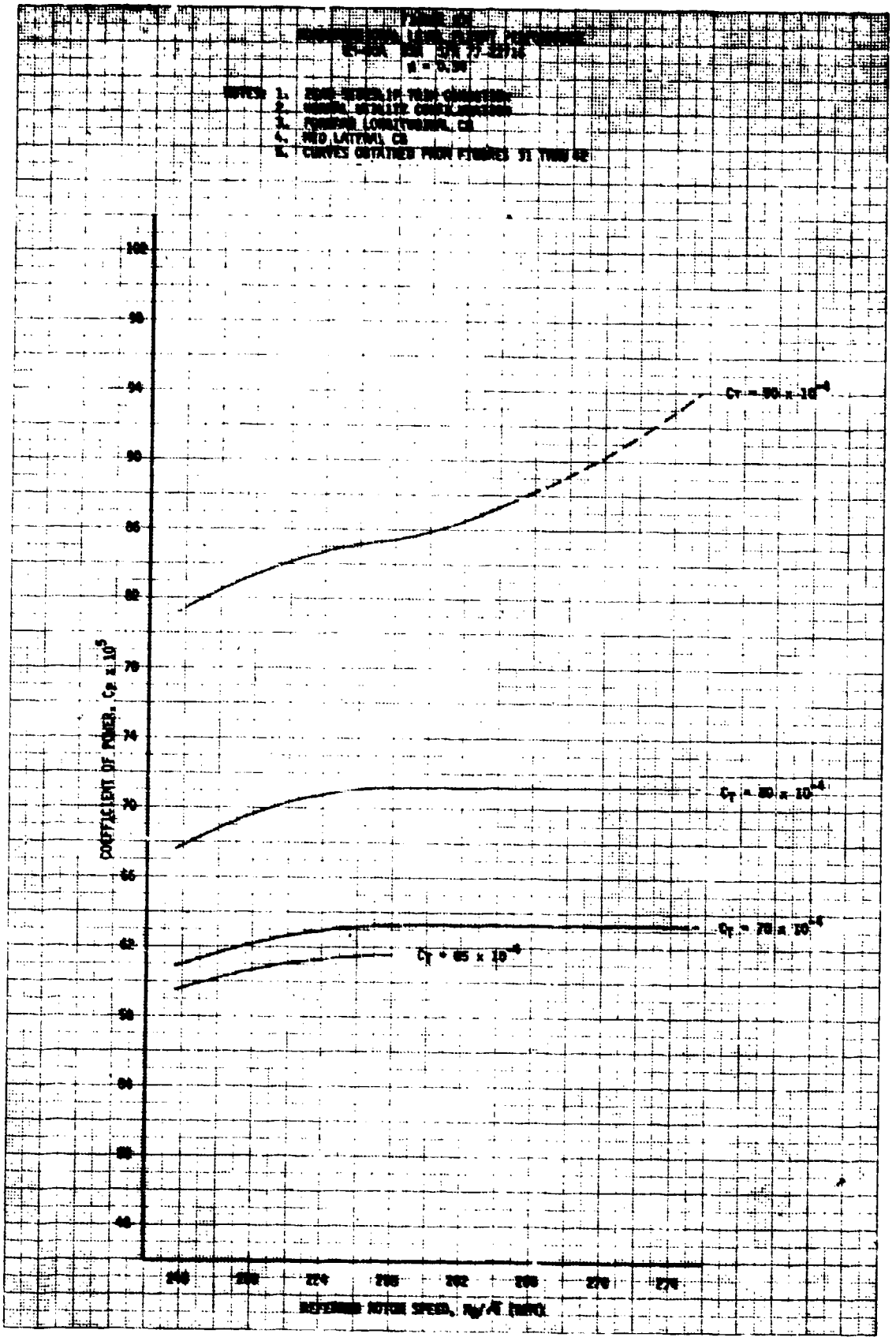


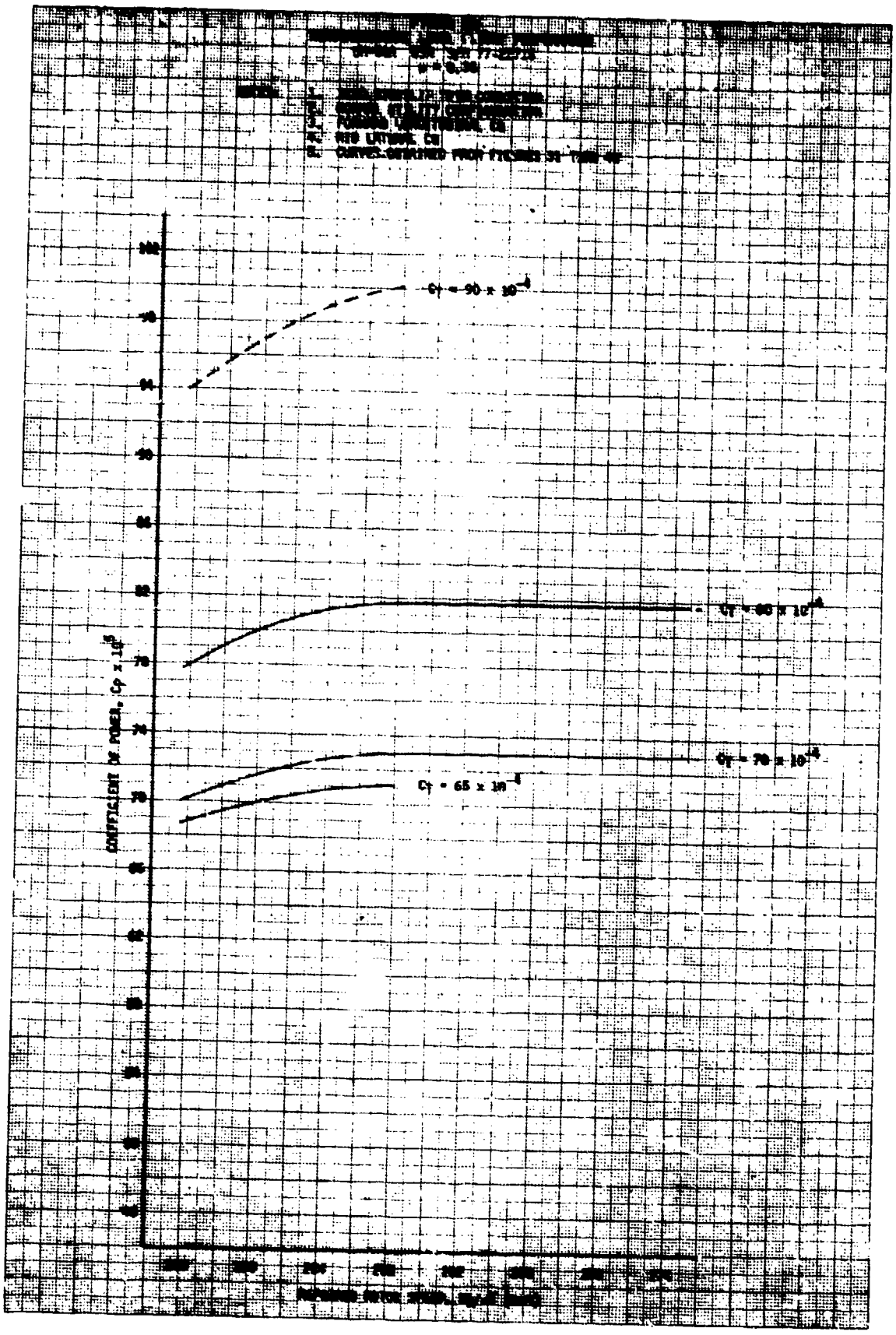

C_T = 110 x 10^{-6}
C_T = 100 x 10^{-6}
C_T = 90 x 10^{-6}
C_T = 80 x 10^{-6}
C_T = 70 x 10^{-6}
C_T = 65 x 10^{-6}
COEFFICIENT OF POWER, C_P x 10^5
246
250
254
258
262
266
270
274


$C_T = 100 \times 10^{-6}$
$C_T = 90 \times 10^{-6}$
$C_T = 80 \times 10^{-6}$
$C_T = 70 \times 10^{-6}$
$C_T = 65 \times 10^{-6}$
COEFFICIENT OF POWER, $C_Q \times 10^5$
246
250
254
258
262
266
270
274
RESIDUAL ROTOR SPEED, Ω_R/Ω (RPM)


$C_T = 100 \times 10^{-6}$
$C_T = 90 \times 10^{-6}$
$C_T = 80 \times 10^{-6}$
$C_T = 70 \times 10^{-6}$
$C_T = 65 \times 10^{-6}$
COEFFICIENT OF POWER, $C_P \times 10^6$


C_T = 100 x 10^{-4}
C_T = 90 x 10^{-4}
C_T = 80 x 10^{-4}
C_T = 70 x 10^{-4}
C_T = 65 x 10^{-4}
COEFFICIENT OF POWER, C_P x 10^5
REFERRED ROTOR SPEED, $\Omega_R/\sqrt{\theta}$ (RPM)
246
250
254
258
262
266
270
274


COEFFICIENT OF POWER, $C_P \times 10^3$
$C_T = 80 \times 10^{-6}$
$C_T = 80 \times 10^{-6}$
$C_T = 70 \times 10^{-6}$
$C_T = 65 \times 10^{-6}$


$C_T = 90 \times 10^{-6}$
$C_T = 80 \times 10^{-6}$
$C_T = 70 \times 10^{-6}$
$C_T = 65 \times 10^{-6}$

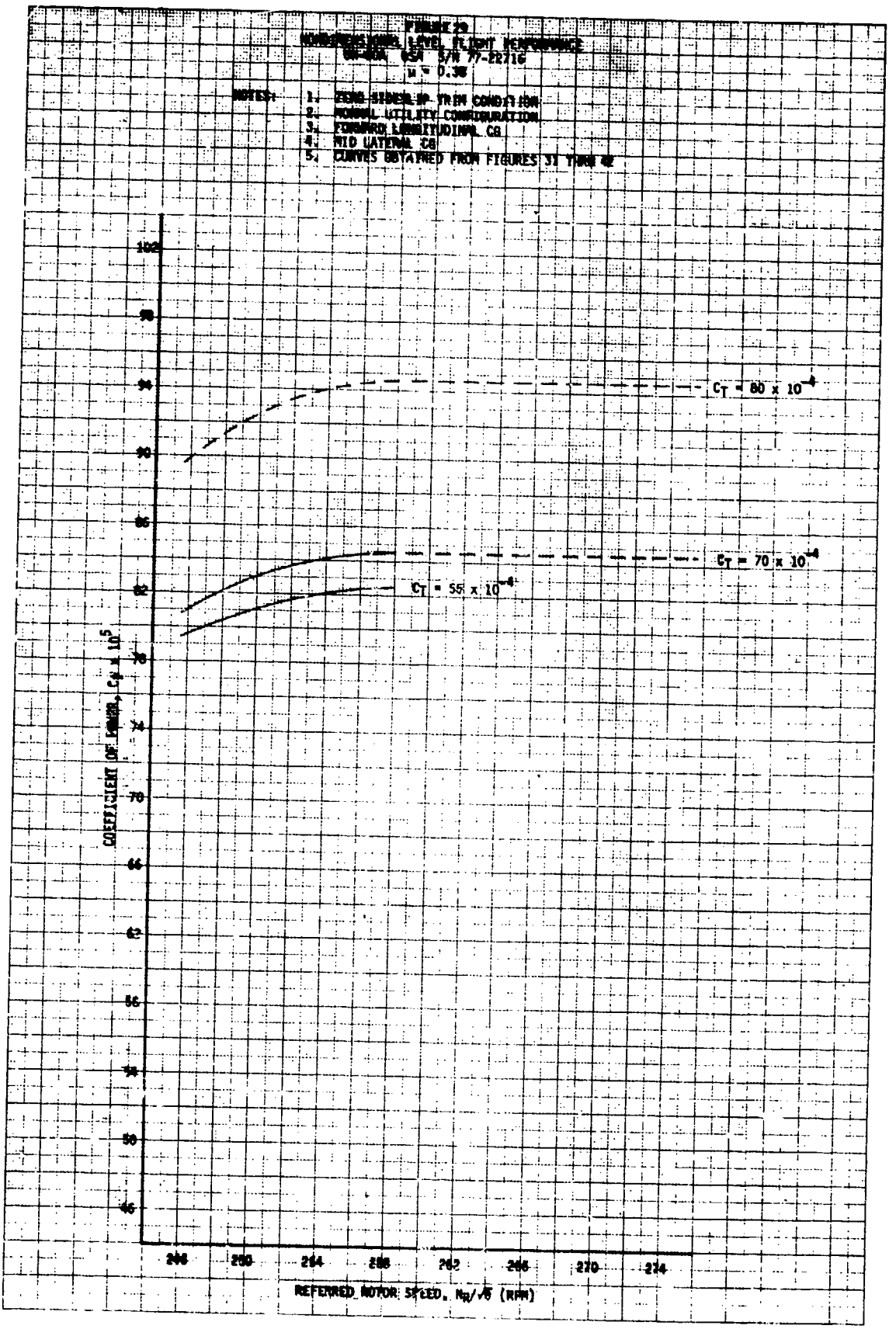


FIGURE 29
NONLINEAR LEVEL FLIGHT PERFORMANCE
SIKORSKY UTTAS S/N 77-22716
μ = 0.38
NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NOMINAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. CURVES OBTAINED FROM FIGURES 31 THRU 40
$C_T = 80 \times 10^{-4}$
$C_T = 70 \times 10^{-4}$
$C_T = 55 \times 10^{-4}$
COEFFICIENT OF POWER, $C_P \times 10^5$
REFERRED ROTOR SPEED, $N_R/\sqrt{\theta}$ (RPM)

FIGURE 30.
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716
$\mu = 0.40$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. CURVES OBTAINED FROM FIGURES 31 THRU 42

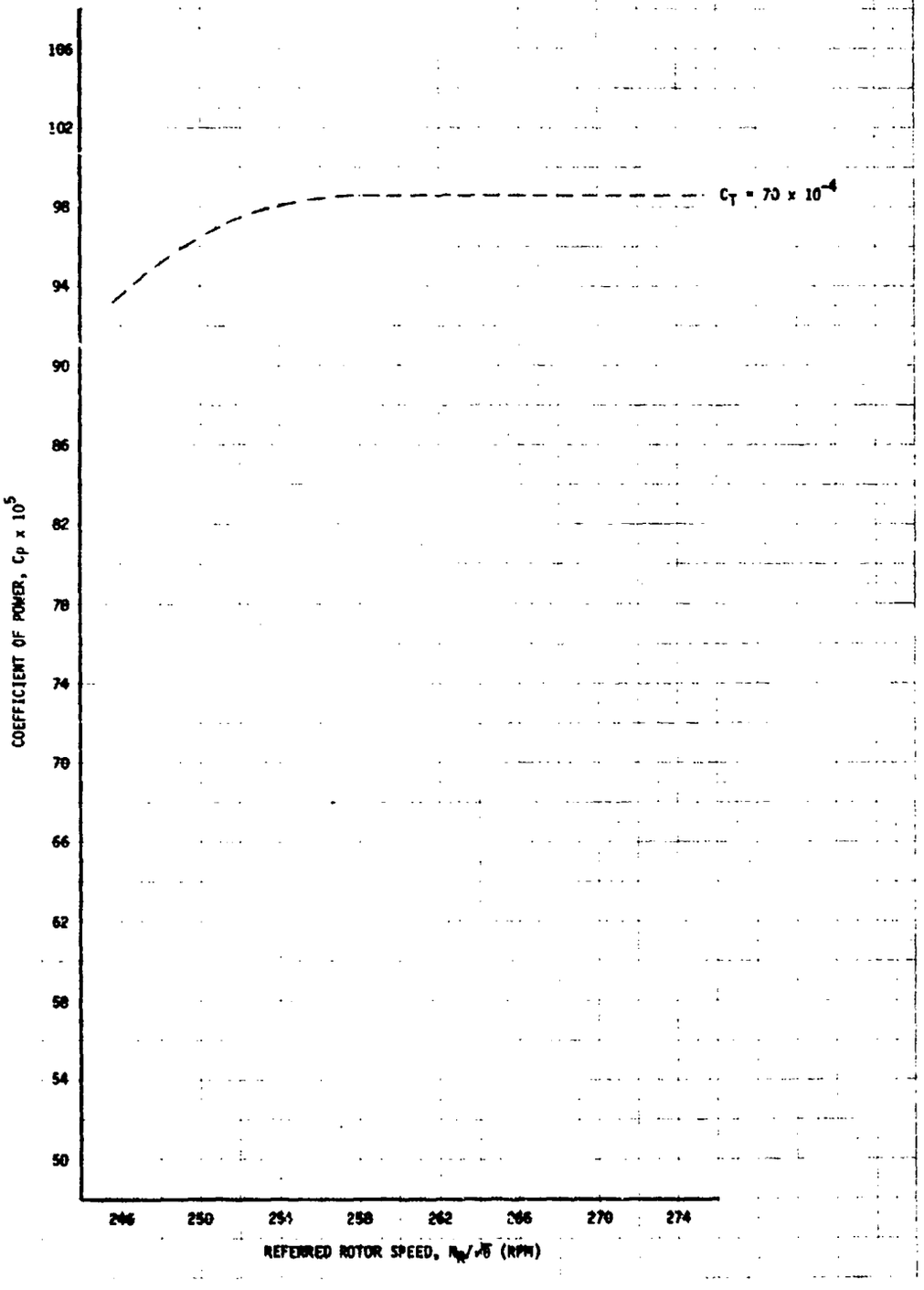


FIGURE 31
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$$\frac{N_R}{\sqrt{\theta}} = 245.6$$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46 THRU 67

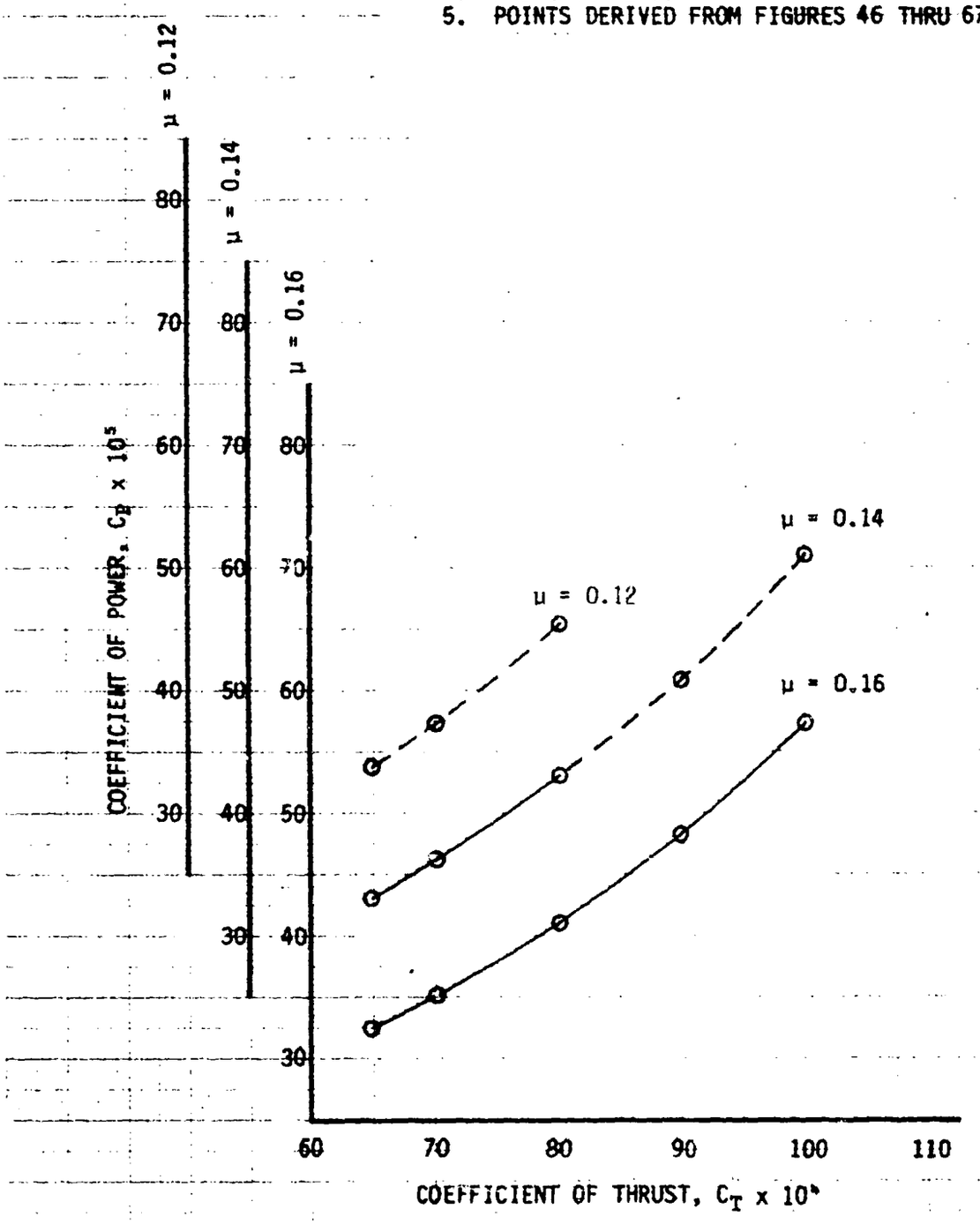


FIGURE 32
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{N_R}{\sqrt{\theta}} = 245.6$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46. THRU 67.

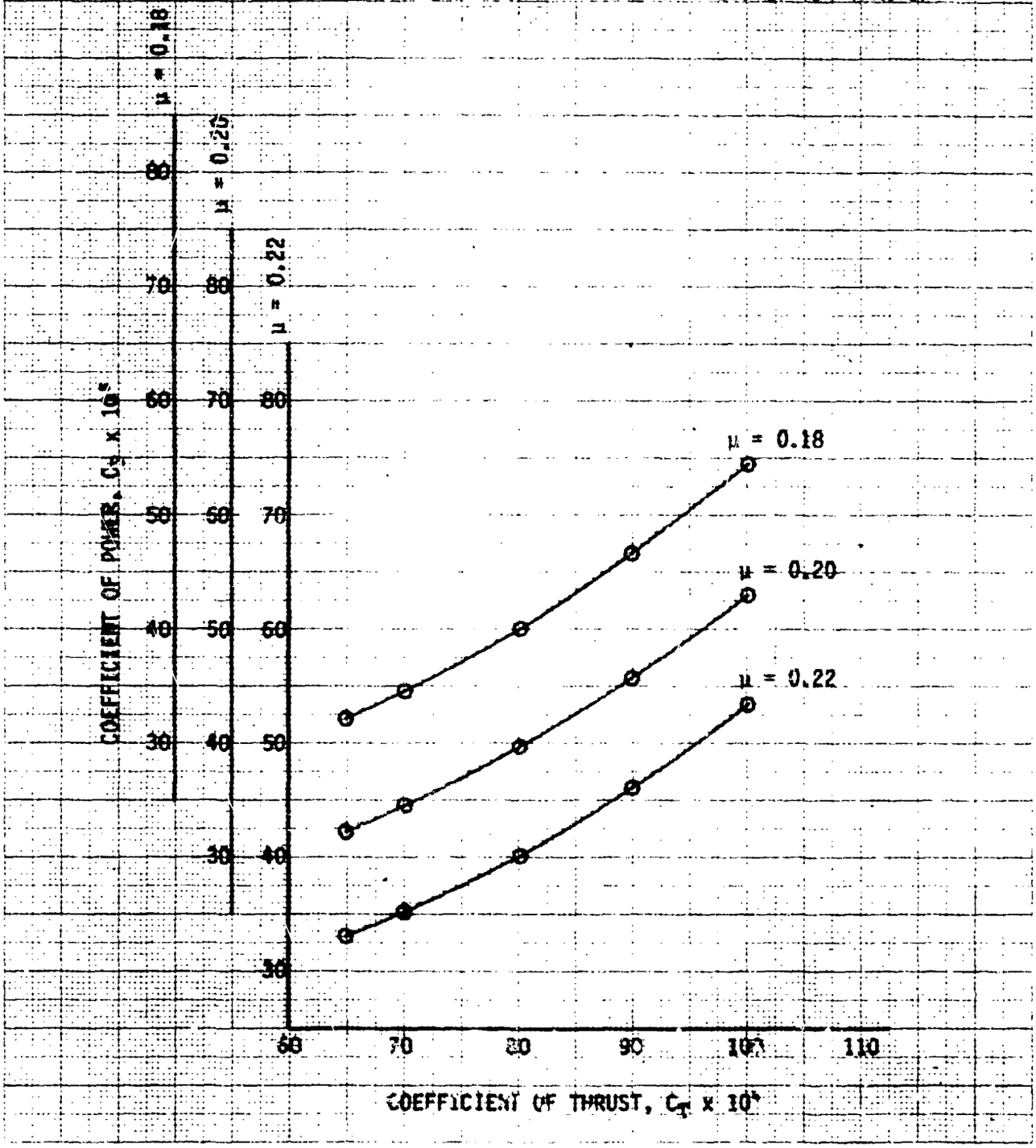


FIGURE 33
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$$\frac{N_R}{\sqrt{\theta}} = 245.6$$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46 THRU 67

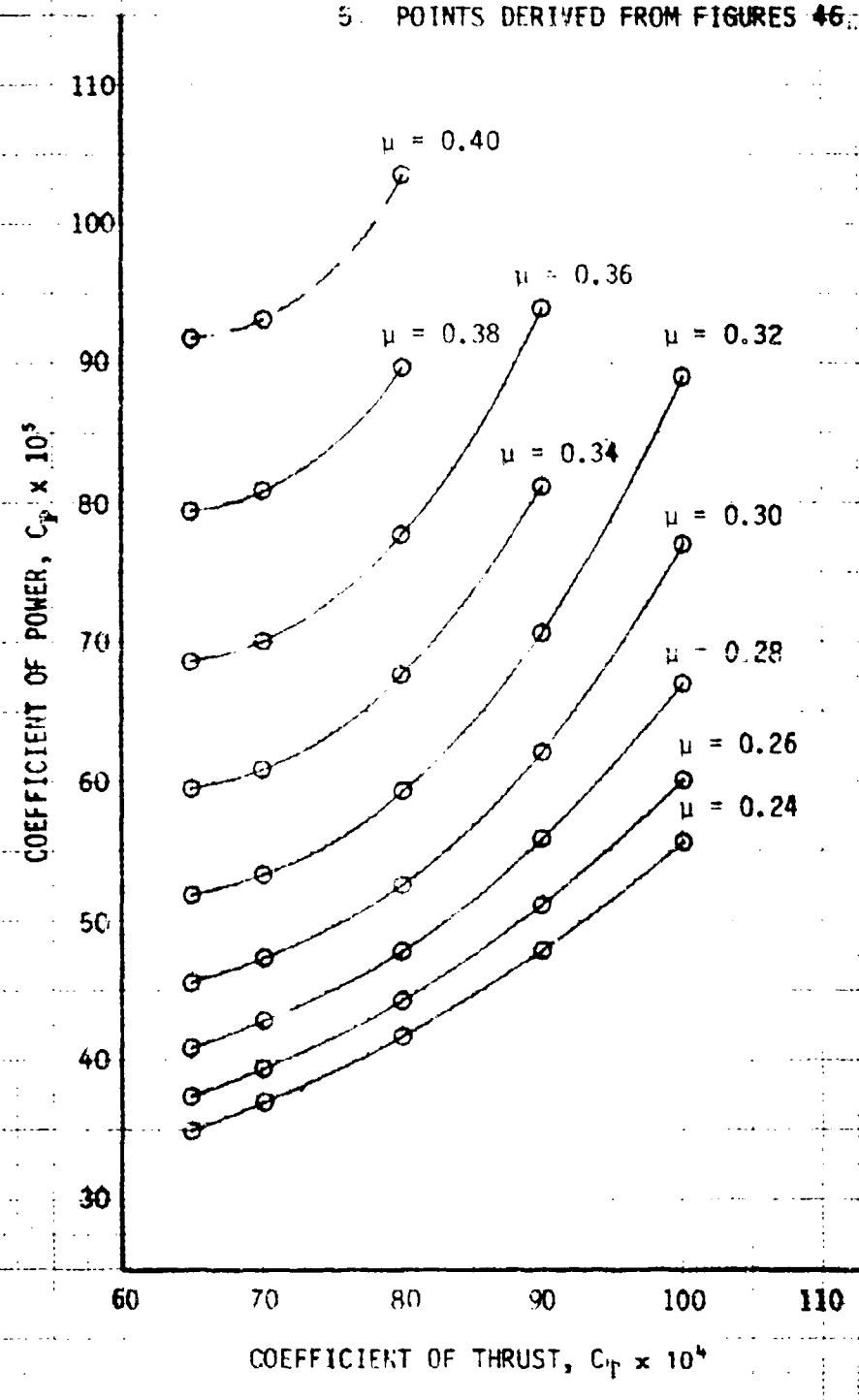


FIGURE 34
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{W_a}{\sigma\theta} = 258.0$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46 THRU 67.

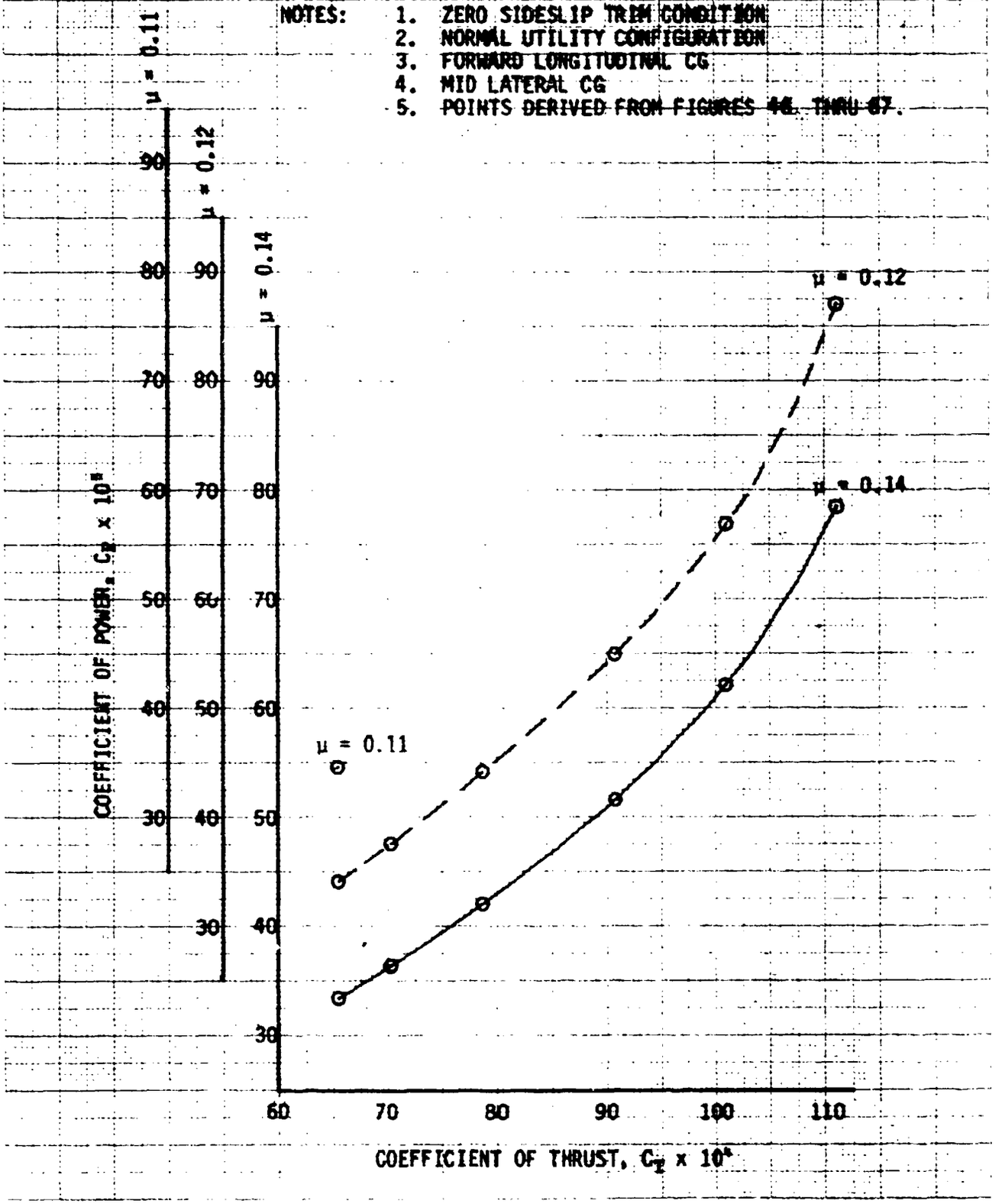


FIGURE 36

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

OH-6CA USN S/N 77-22716

N_R

$\sqrt{\theta}$ = 258.0

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NOMINAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46 THRU 67.

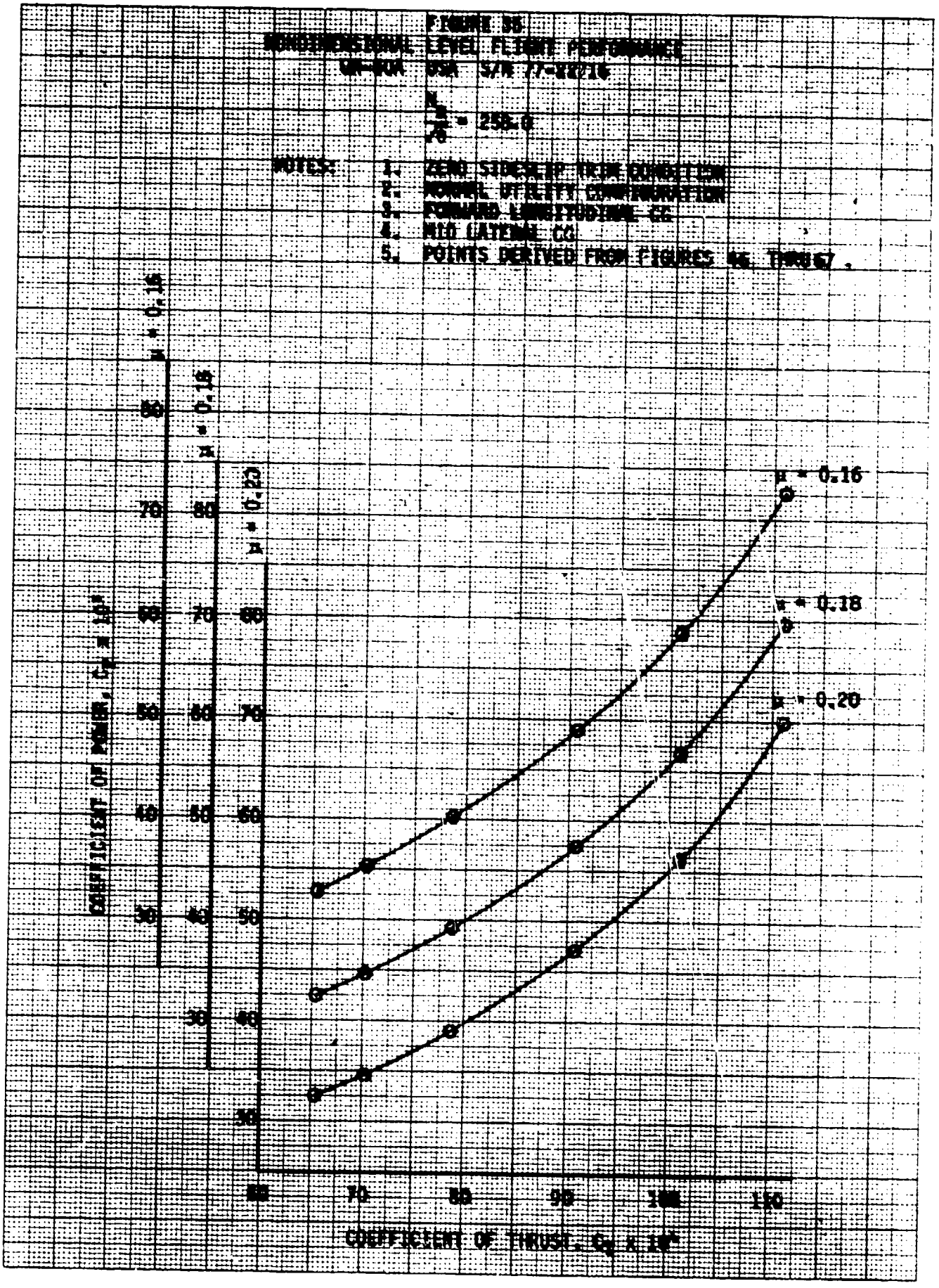


FIGURE 36
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{N_R}{\sqrt{\theta}}$ = 258.0

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46. THRU 67.

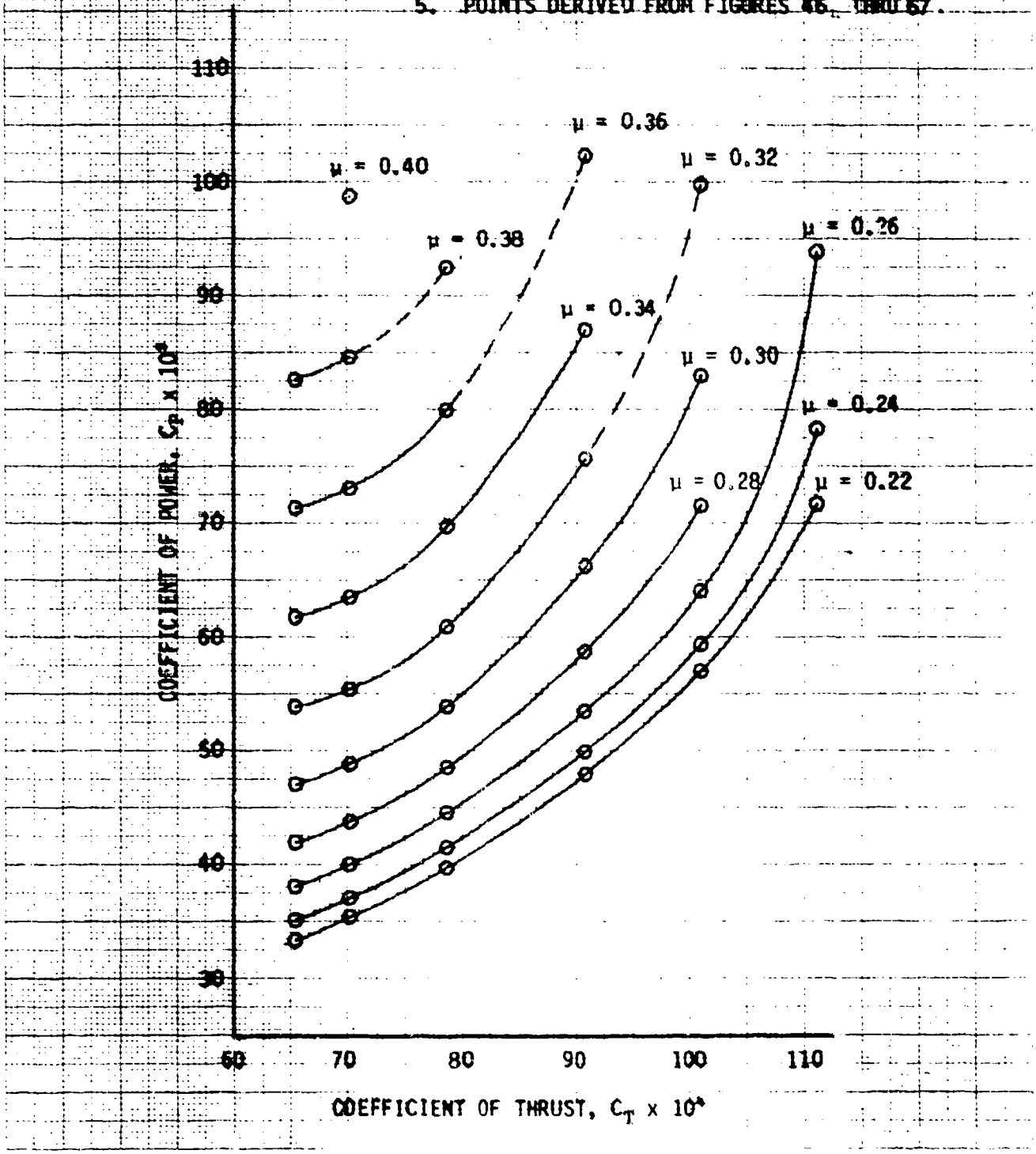


FIGURE 37
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{N_R}{\sqrt{\theta}} = 265.0$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46. THRU 67.

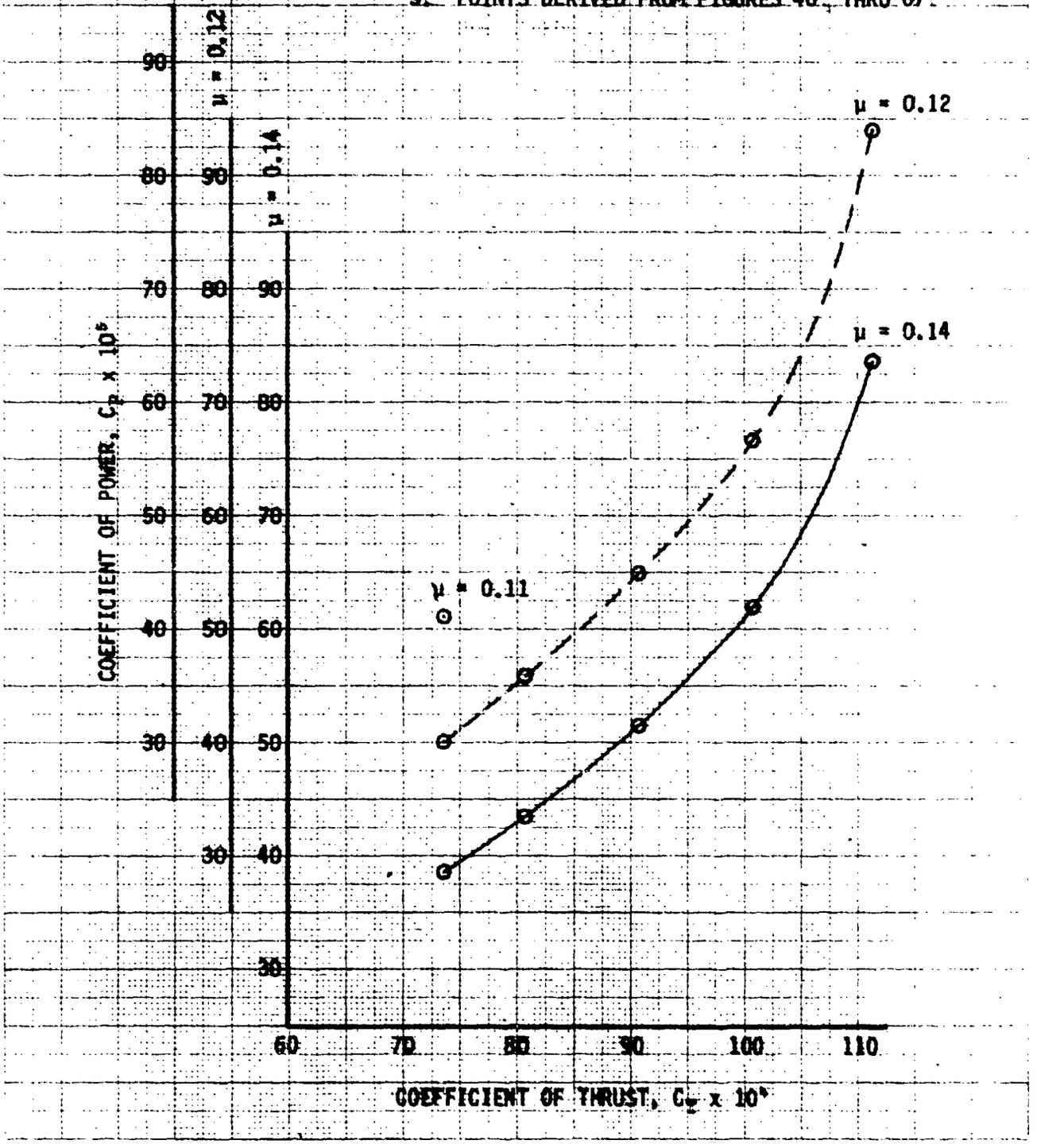


FIGURE 38
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{N_R}{\sqrt{\theta}} = 265.0$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES #6 THRU 67.

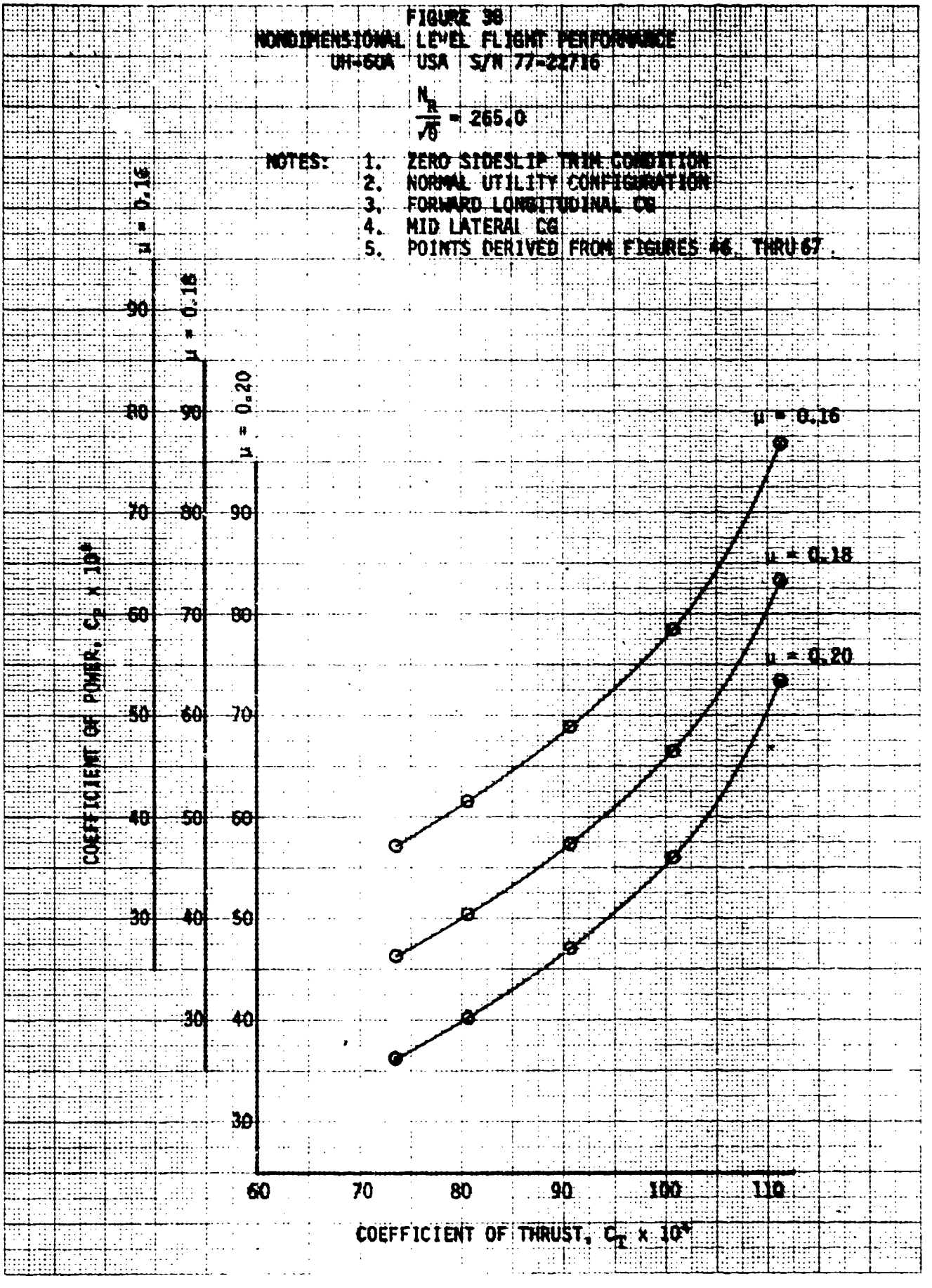


FIGURE 39
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{N_R}{\sqrt{\theta}} = 265.0$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46 THRU 67.

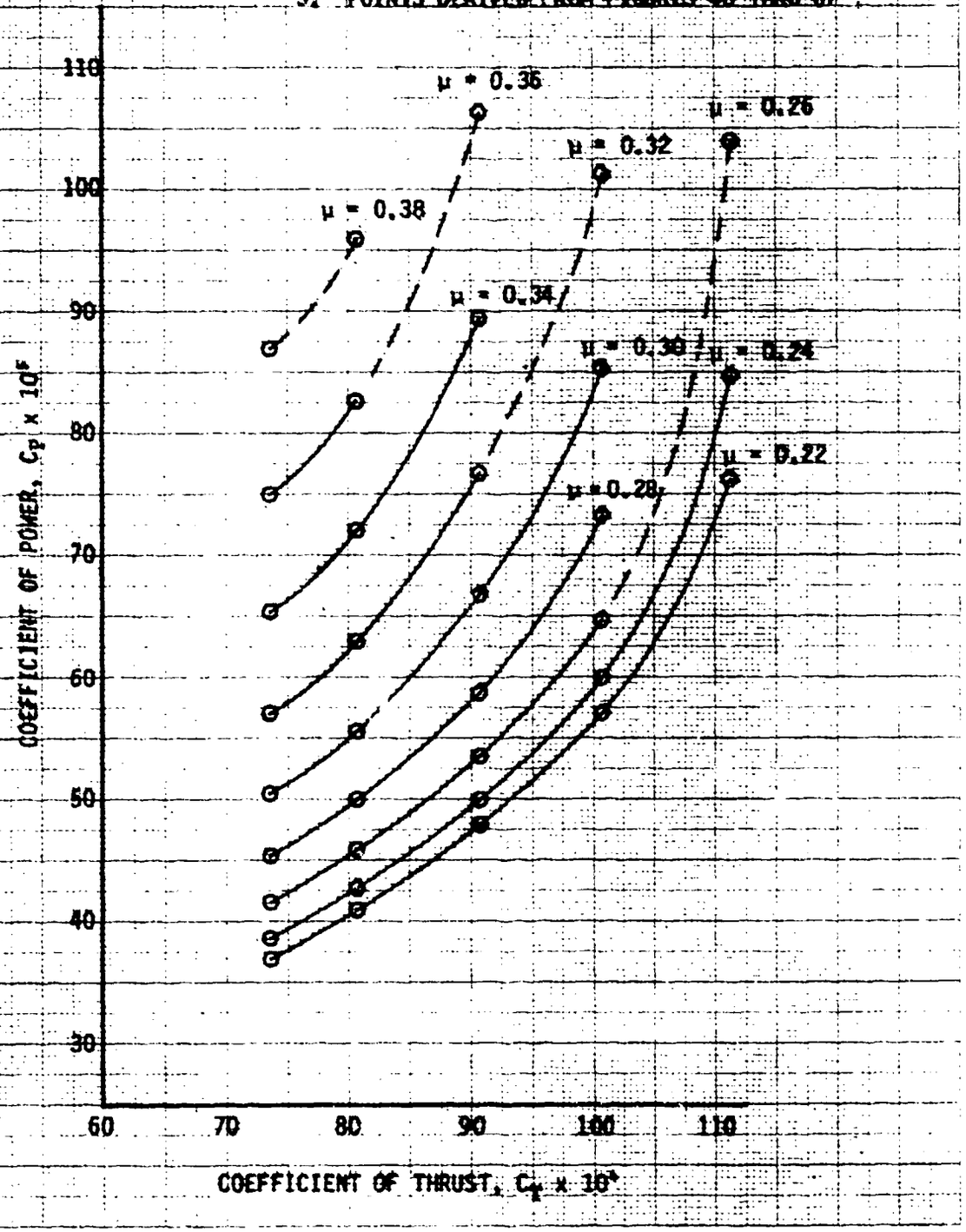
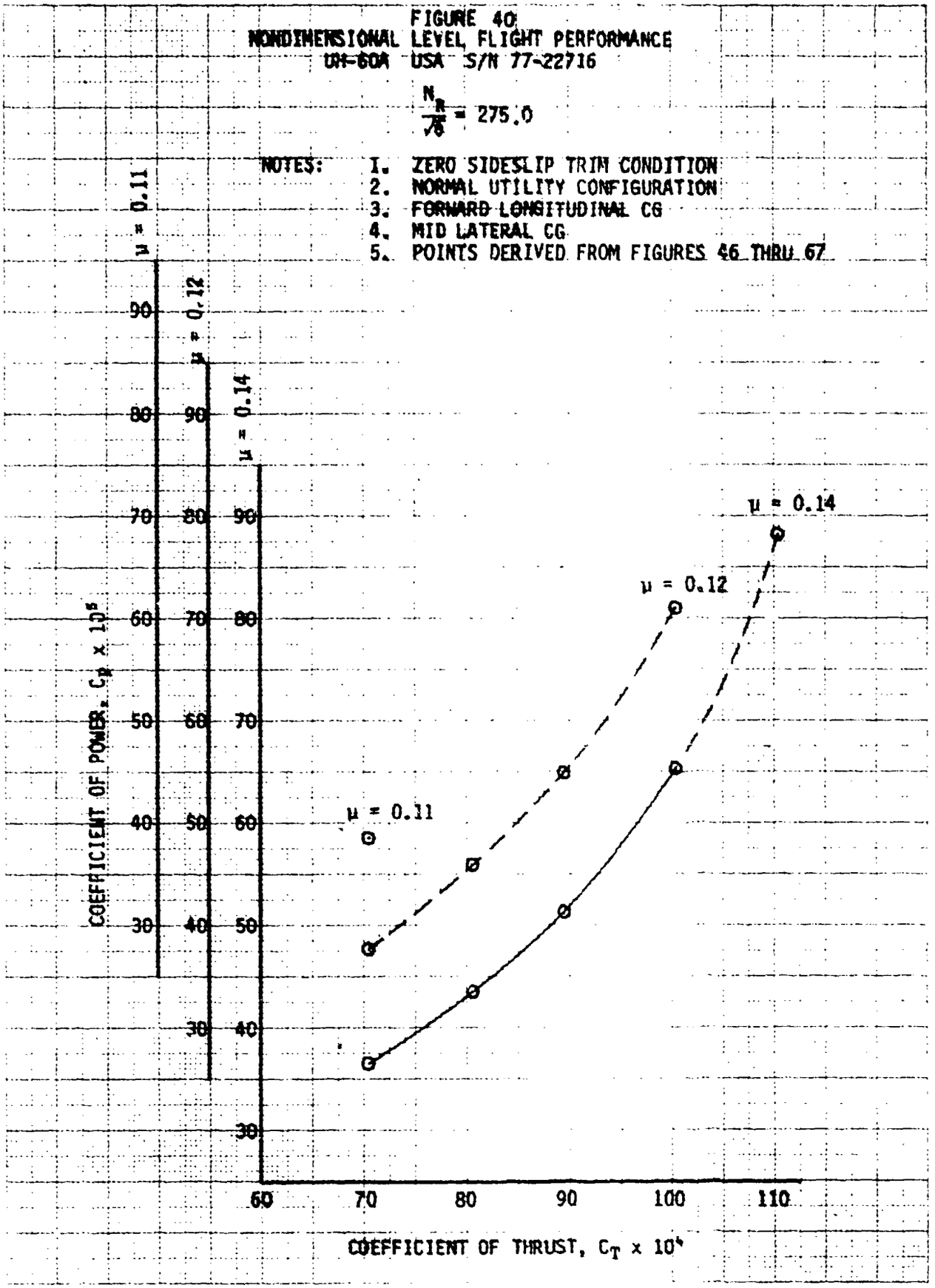


FIGURE 40
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{N_R}{\sqrt{\theta}} = 275.0$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46 THRU 67



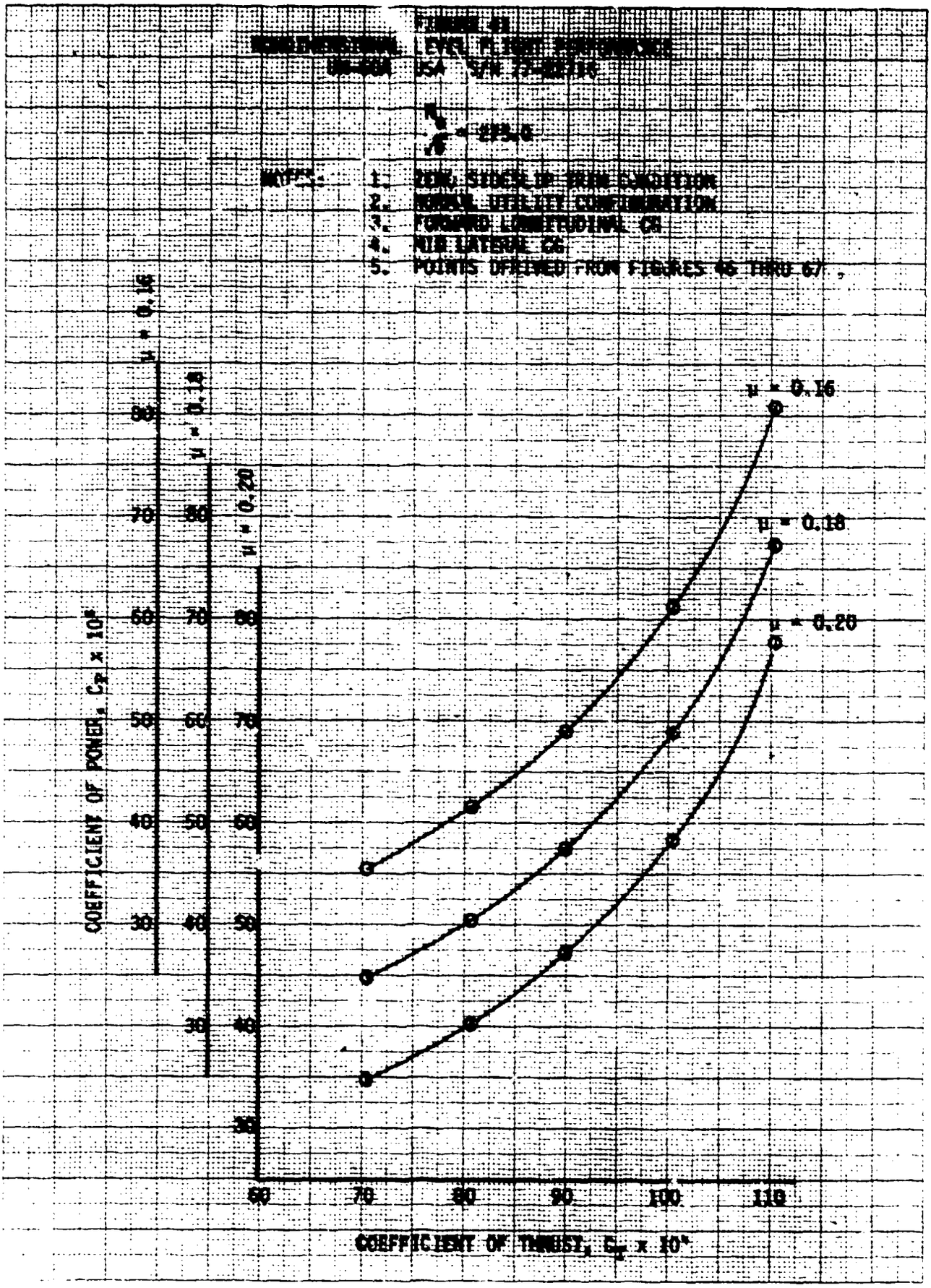

1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. AFT LATERAL CG
5. POINTS DEFINED FROM FIGURES 46 THRU 67
μ = 0.16
μ = 0.18
μ = 0.20
COEFFICIENT OF POWER, $C_P \times 10^4$
COEFFICIENT OF THRUST, $C_T \times 10^4$
30
40
50
60
70
80
90
100
110

FIGURE 42
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-60A USA S/N 77-22716

$\frac{N_R}{\sqrt{\theta}} = 275.0$

NOTES:
1. ZERO SIDESLIP TRIM CONDITION
2. NORMAL UTILITY CONFIGURATION
3. FORWARD LONGITUDINAL CG
4. MID LATERAL CG
5. POINTS DERIVED FROM FIGURES 46 THRU 67

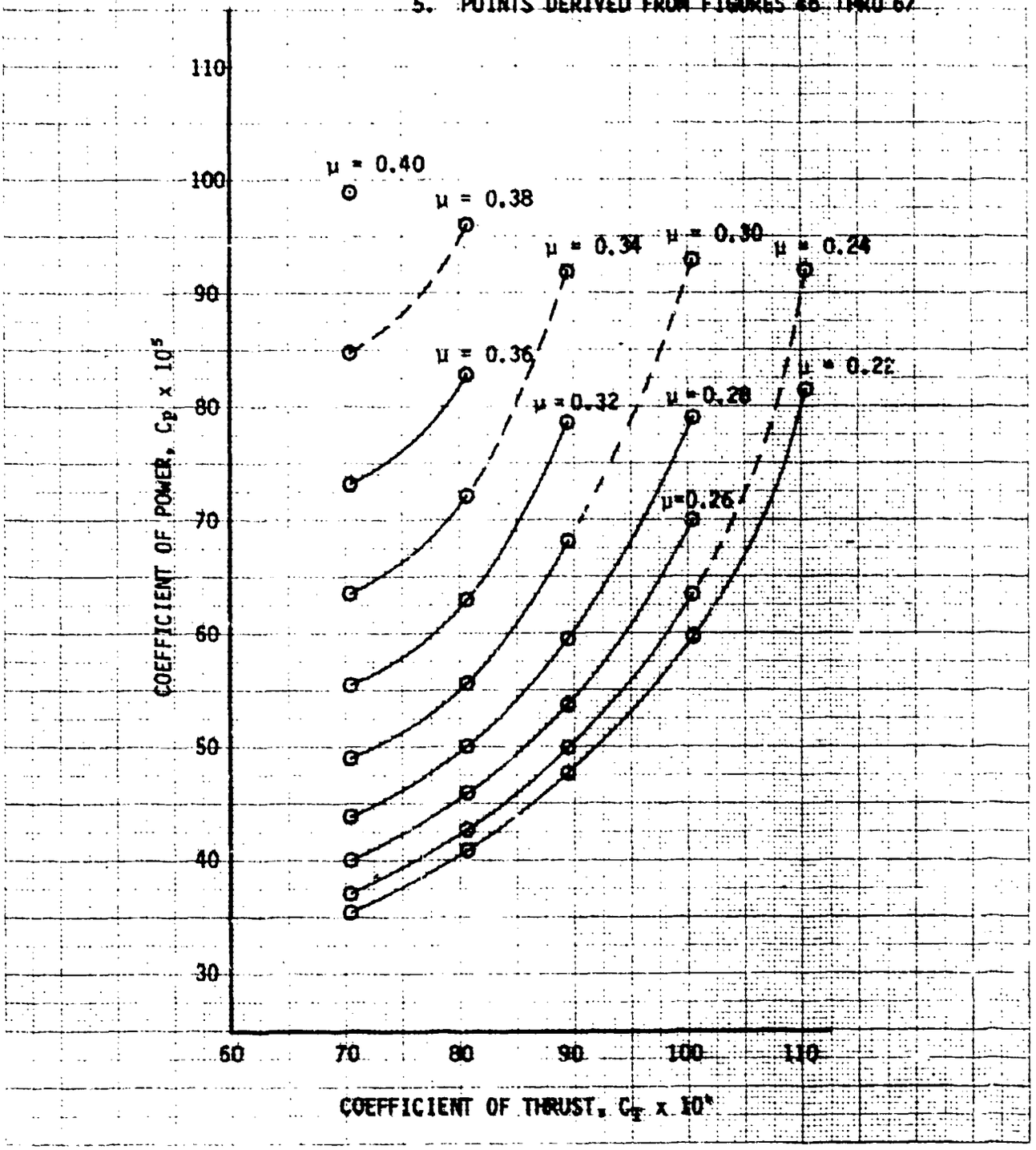


FIGURE 43
INHERENT SIDESLIP
UH-60A USA S/N 77-22718

NOTES: 1. NORMAL UTILITY CONFIGURATION
2. FORWARD LONGITUDINAL CG
3. MID LATERAL CG
4. AVERAGE REFERRED ROTOR SPEED = 245.6 THRU 275.6 RPM
5. CURVES DERIVED FROM FIGURES 47 THRU 67

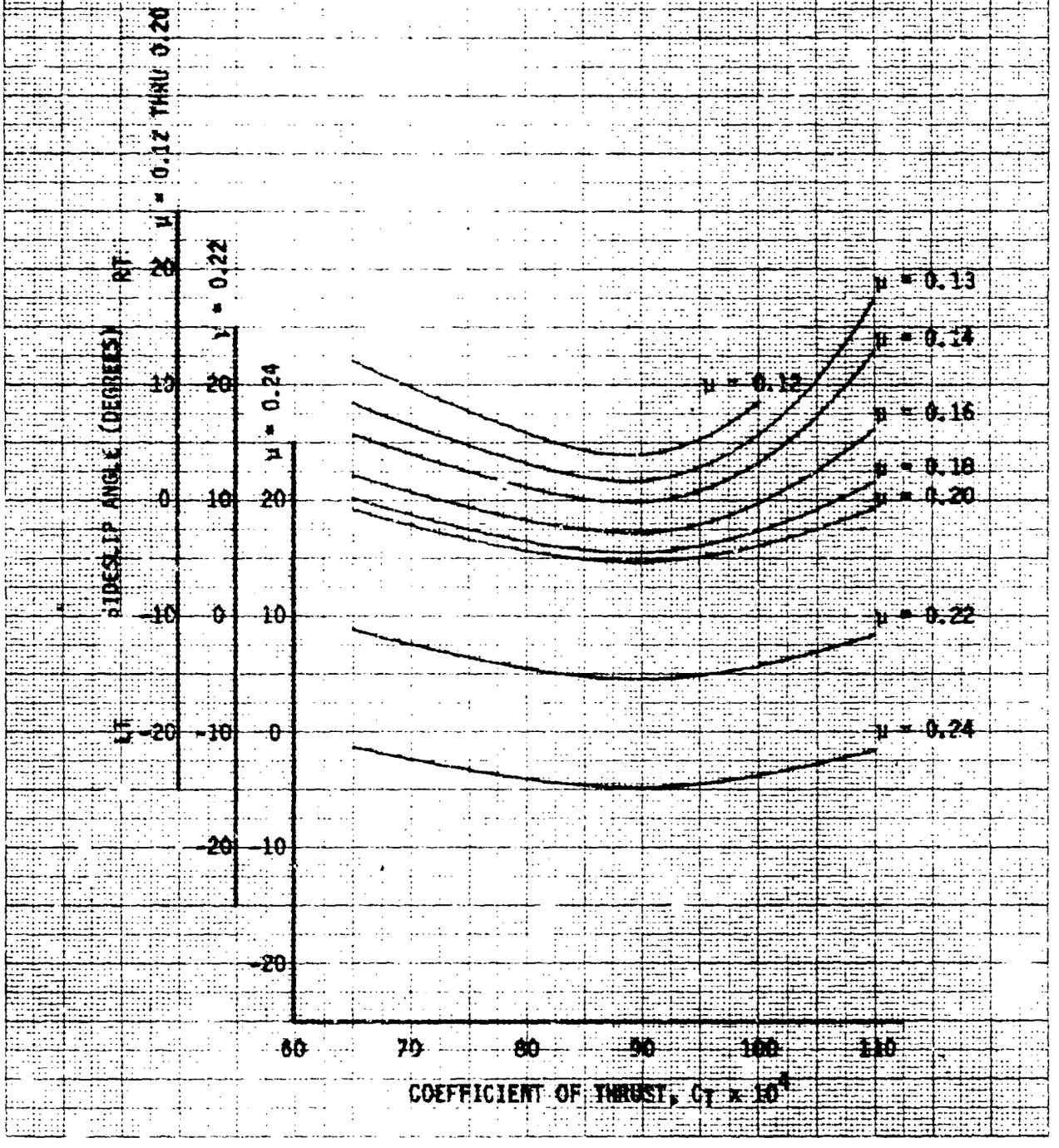


FIGURE 44
INHERENT SIDESLIP
UH-60A USA S/N 77-22716

NOTES: 1. NORMAL UTILITY CONFIGURATION
2. FORWARD LONGITUDINAL CG
3. MID LATERAL CG
4. AVERAGE REFERRED ROTOR SPEED = 245.6 THRU 275.6 RPM
5. CURVES DERIVED FROM FIGURES 47 THRU 67

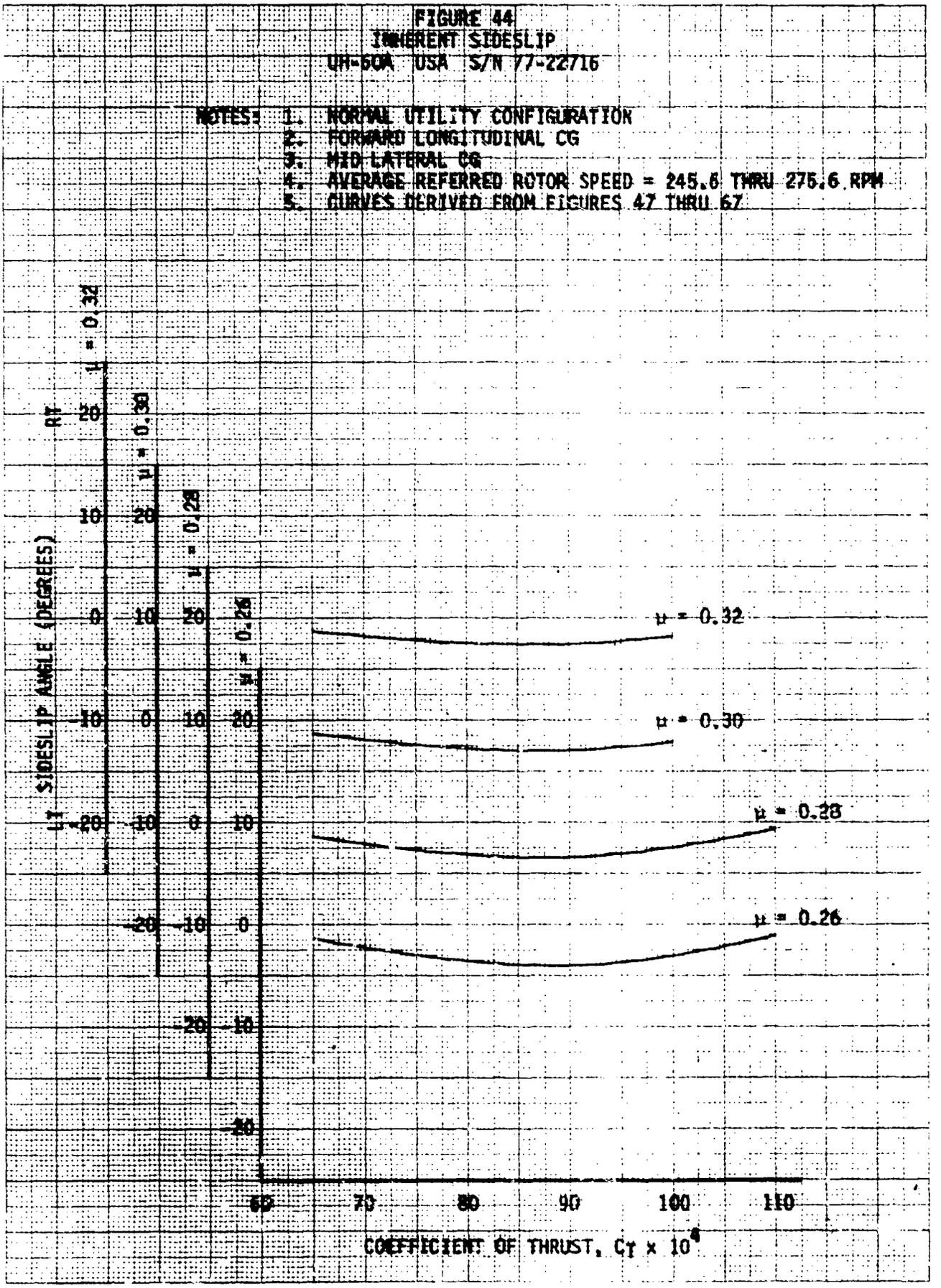


FIGURE #5
INHERENT SIDESLIP
UH-60A USA S/N 77-22716

NOTES:
1. NORMAL UTILITY CONFIGURATION
2. FORWARD LONGITUDINAL CG
3. MID LATERAL CG
4. AVERAGE REFERRED ROTOR SPEED = 245.6 THRU 275.6 RPM
5. CURVES DERIVED FROM FIGURES 47 THRU 67

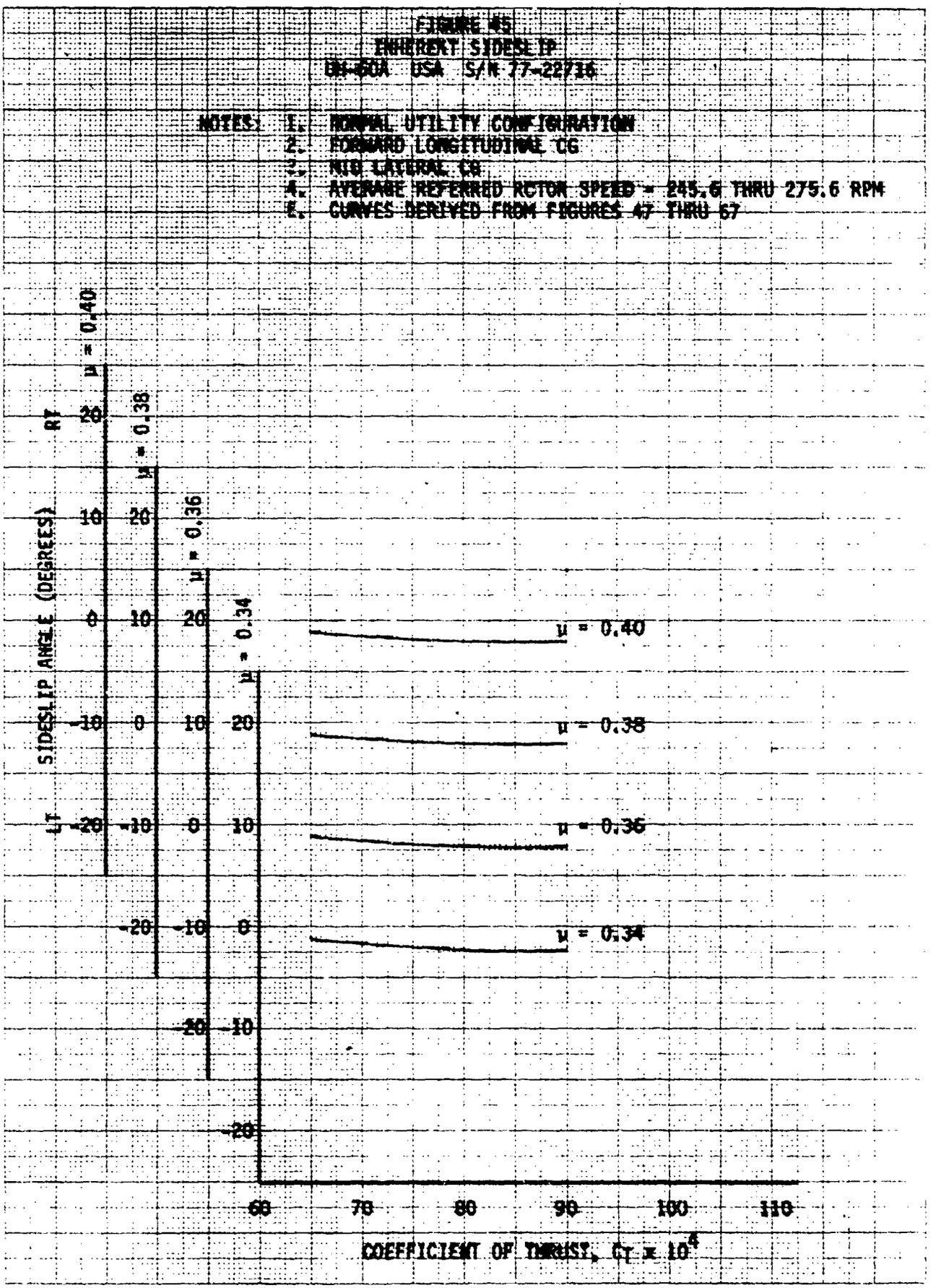


FIGURE 46

CHANGE IN EQUIVALENT FLAT-PLATE AREA WITH SIDESLIP

UH-60A USA S/N 77-22716

SYMBOL	AVG TRUE AIRSPEED (KT)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG REFERRED ROTOR SPEED (RPM)	AVG C_T ($\times 10^4$)	CONFIGURATION
○	85	346.9	0.0	258	70.19	NORMAL UTILITY
◇	141	346.9	0.0	259	70.17	NORMAL UTILITY
□	86	347.2	0.0	258	90.51	NORMAL UTILITY
▽	122	347.0	0.0	258	90.52	NORMAL UTILITY
△	136	347.1	0.0	258	90.68	NORMAL UTILITY
⬭	87	347.0	0.0	254	93.42	NORMAL UTILITY
⬠	88	346.9	0.0	271	82.22	NORMAL UTILITY

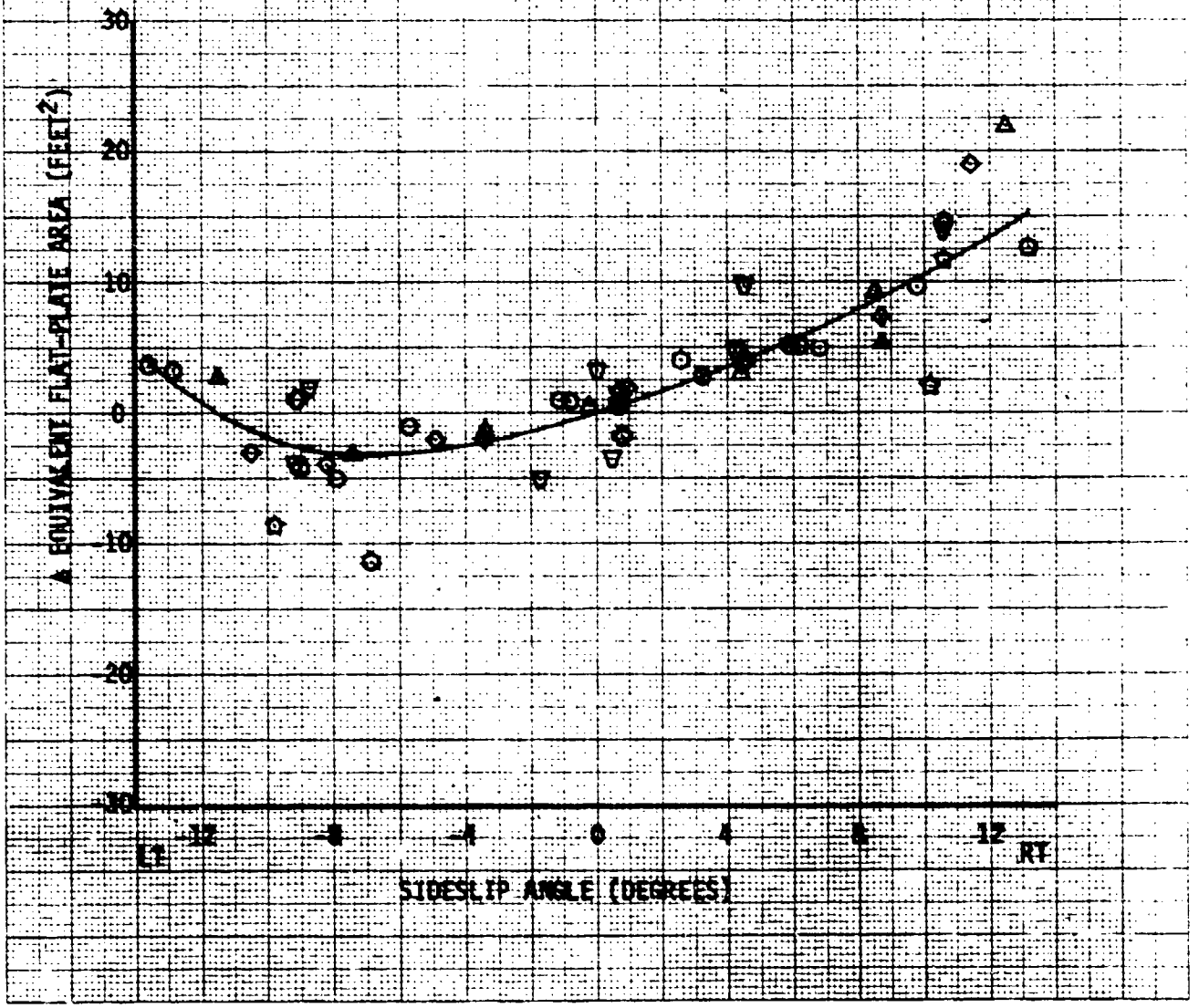


FIGURE 47

LEVEL FLIGHT PERFORMANCE

UH-60A S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG GT	CONFIGURATION
15500	346.9 (FWD)	0.2 (LT)	2060	12.0	245.7	0.000	60% UTIL

NOTE: BALL CENTER TRIM CONDITION

SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)

0.25
0.20
0.15
0.10
0.05
0.00

CURVE BASED ON SPECIFICATION FUEL FLOW

CURVE OBTAINED FROM FIGS. 45 THRU 45

MCP

CURVE DERIVED FROM FIGS. 15 THRU 30 AND 43 THRU 46

ENGINE SHAFT HORSEPOWER REQUIRED

3200
2800
2400
2000
1600
1200
800

40 60 80 100 120 140 160 180

TRUE AIRSPEED (KNOTS)

PITCH ATTITUDE (DEG)

2°
0
-2°

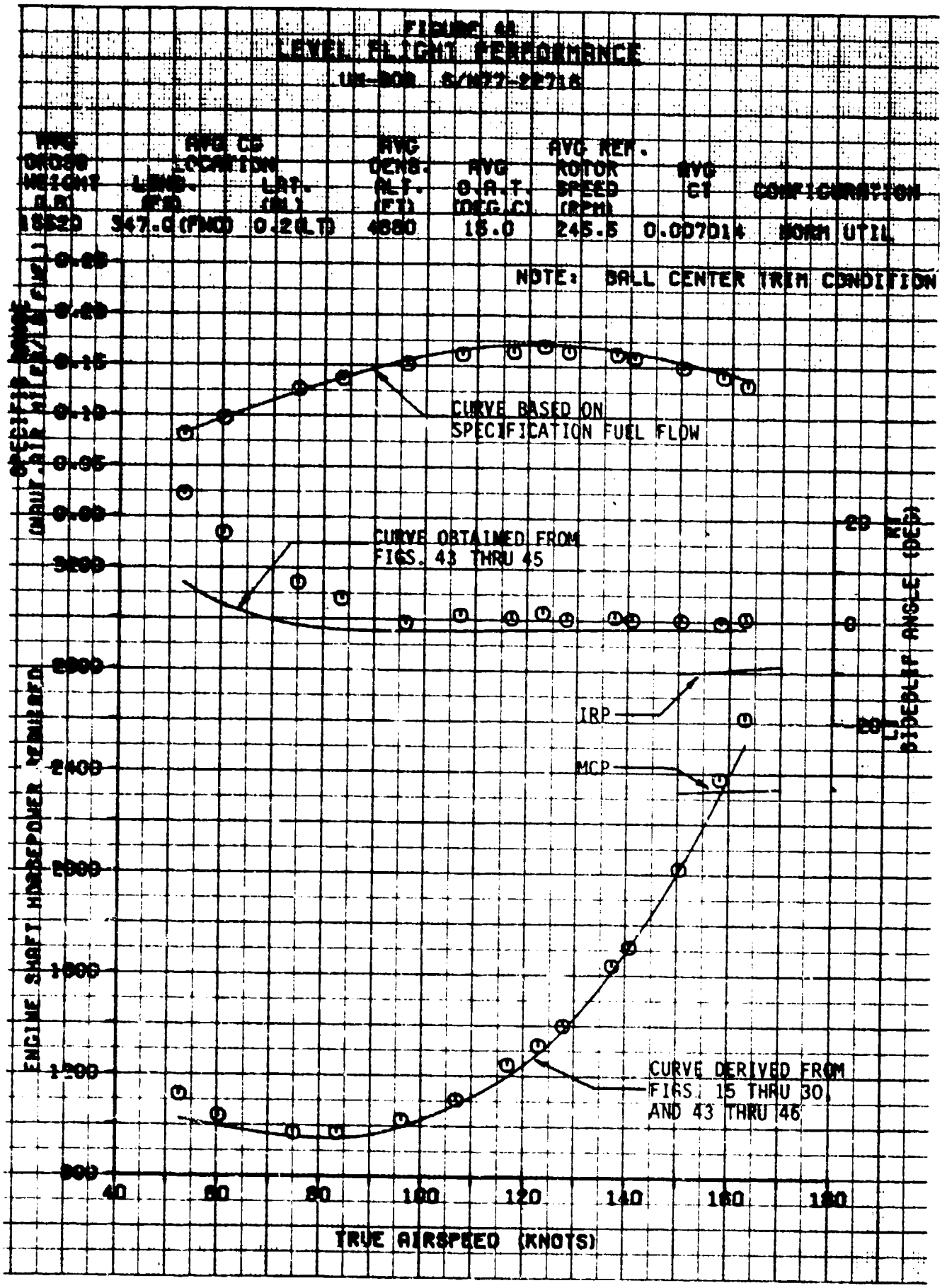
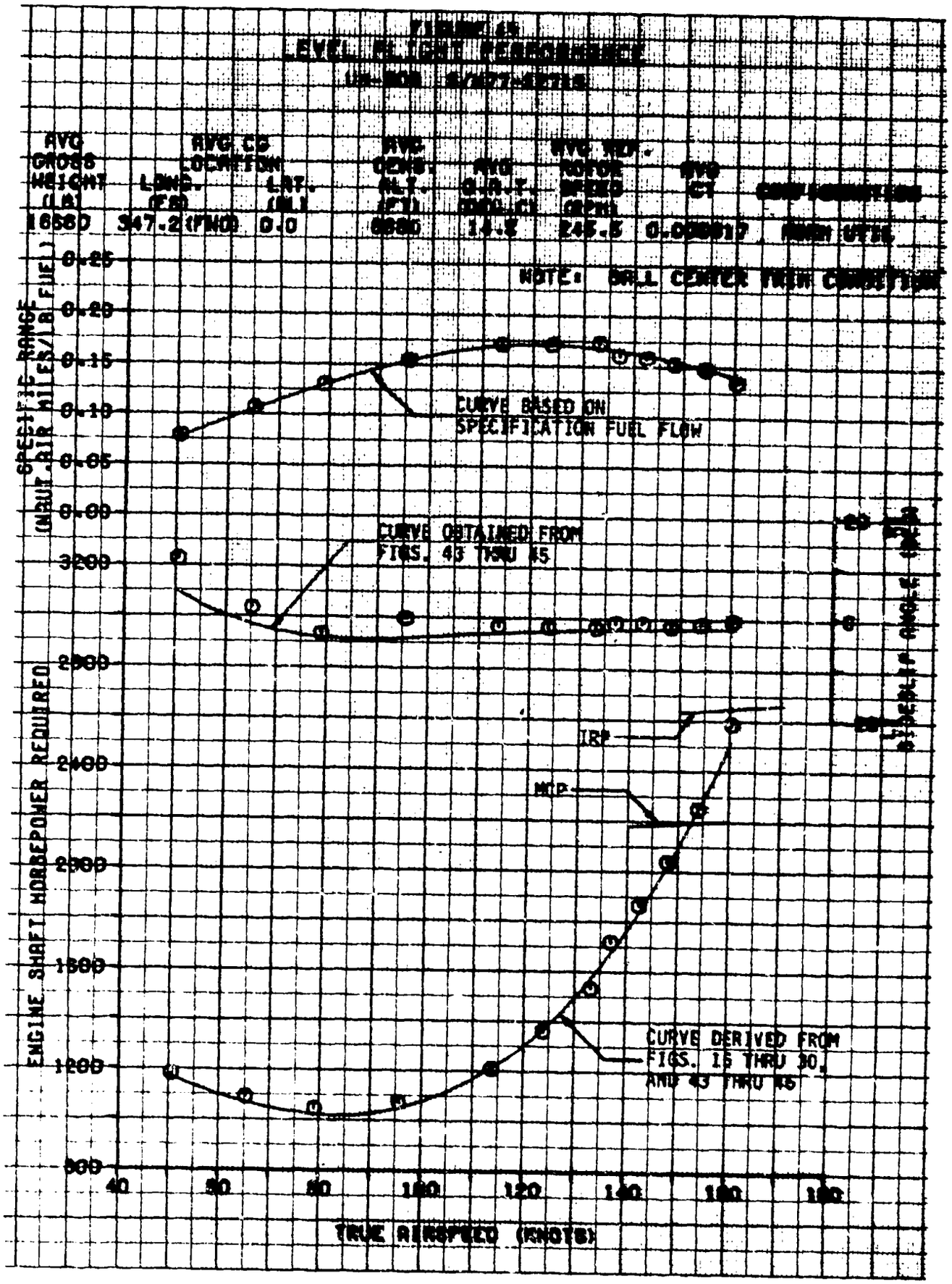


FIGURE 44
LEVEL FLIGHT PERFORMANCE
AVG CG LOCATION
AVG DENS. ALT. (FT)
AVG O.A.T. (DEG.C)
AVG REF. ROTOR SPEED (RPM)
AVG GT
CONFIGURATION
NORM UTIL
NOTE: BALL CENTER TRIM CONDITION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
ENGINE SHAFT HORSEPOWER REQUIRED
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180


AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG. (FS)
LAT. (BL)
AVG DENS. ALT. (FT)
AVG O.A.T. (DEG C)
AVG NOM. ROTOR SPEED (RPM)
AVG SHP
1650 347.2 FWD 0.0 6000 14.2 245.5 0.000017
NOTE: DRILL CENTER FROM CONDITION
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 45
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

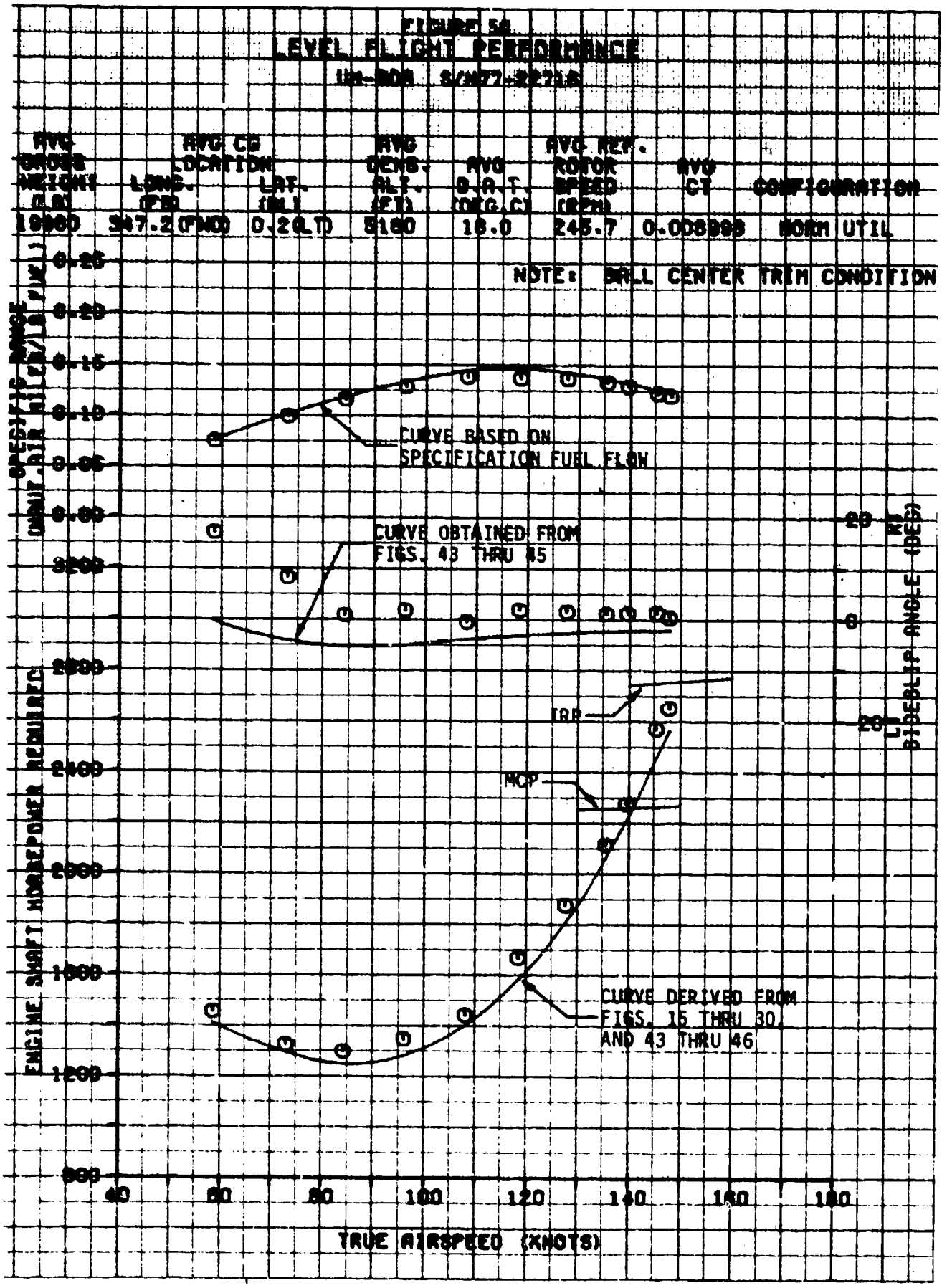


FIGURE 54
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22718
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG. (FS)
LAT. (BL)
AVG DENS. ALT. (FT)
AVG O.A.T. (DEG.C)
AVG REF. ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
19000 347.2 (FWD) 0.2 (LT) 5160 18.0 245.7 0.008988 NORM UTIL
NOTE: BALL CENTER TRIM CONDITION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30 AND 43 THRU 46
SPECIFIC POWER (ENGINE SHAFT HORSEPOWER/LB/HR/LB AIR)
ENGINE SHAFT HORSEPOWER REQUIRED
LT SIDESLIP ANGLE (DEG)
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180

FIGURE 51

LEVEL FLIGHT PERFORMANCE

UH-60A S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	AVG CG LOCATION LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG GI	CONFIGURATION
19150	347.2 (FWD)	0.2 (LT)	8600	18.0	245.6	0.010008	NORM UTIL

NOTE: BALL CENTER TRIM CONDITION

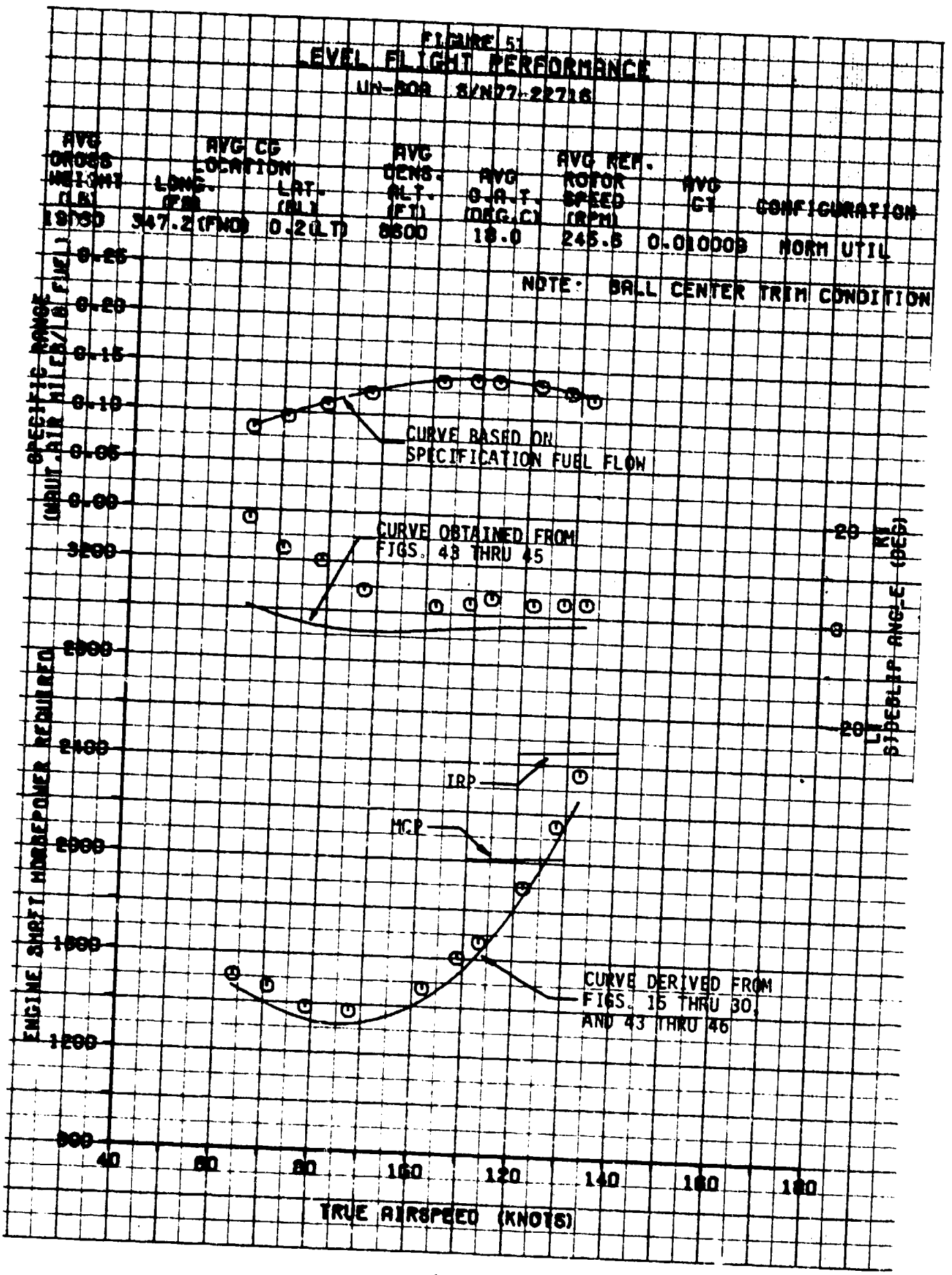


FIGURE 52
LEVEL FLIGHT PERFORMANCE
UH-60A S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
16560	347.3(FWD)	0.0	2660	6.0	258.1	0.006545	NORM UTIL

NOTE: BALL CENTER TRIM CONDITION

CURVE BASED ON SPECIFICATION FUEL FLOW

CURVE OBTAINED FROM FIGS. 43 THRU 45

CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 45

SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)

0.25
0.20
0.15
0.10
0.05
0.00

ENGINE SHAFT HORSEPOWER REQUIRED

3200
2800
2400
2000
1600
1200
800

PITCH ATTITUDE (DEG)

20
0

40 60 80 100 120 140 160 180

TRUE AIRSPEED (KNOTS)

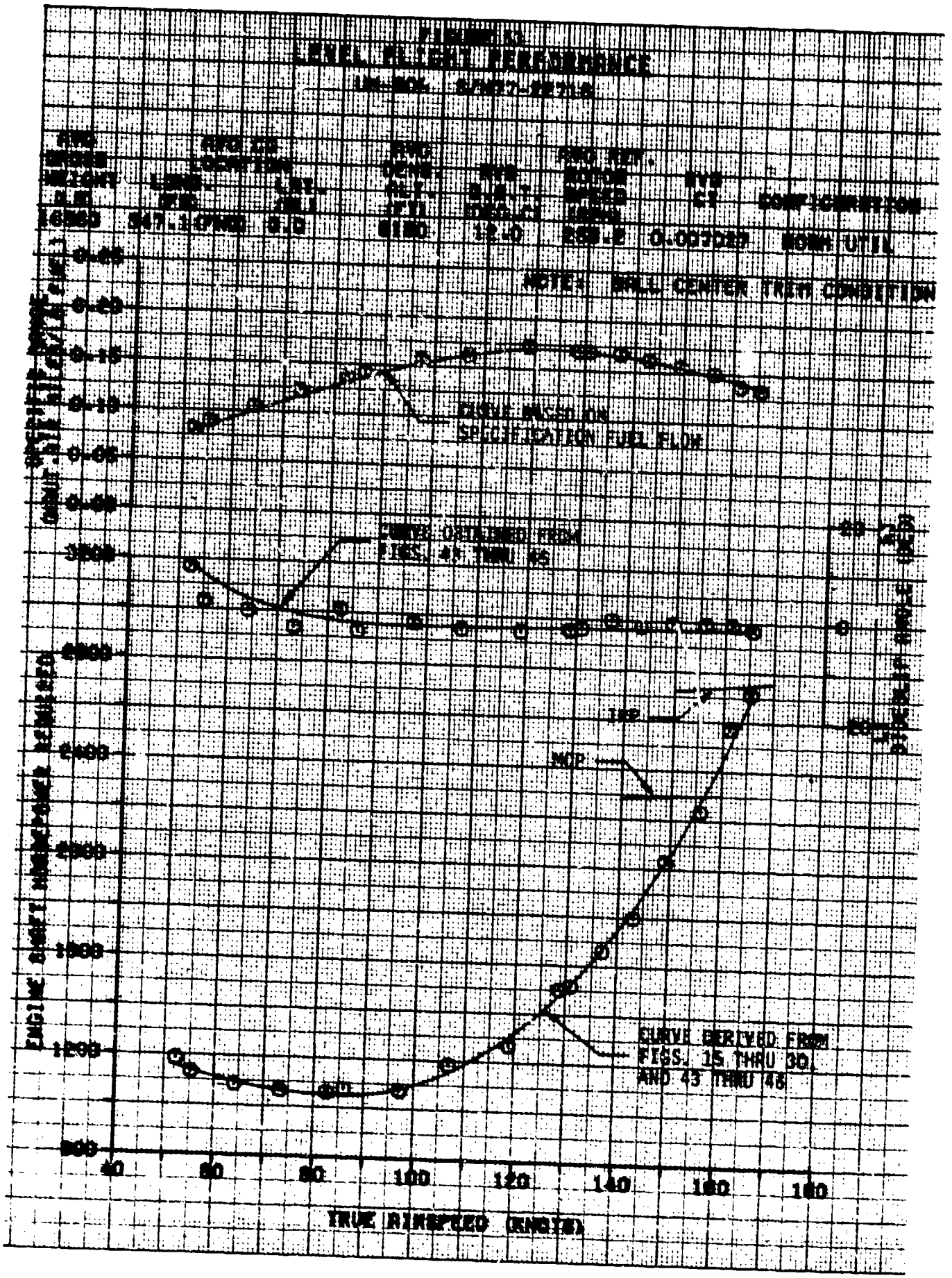


FIGURE 14
FUEL FLOW
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
MOP
TRUE AIRSPEED (KNOTS)

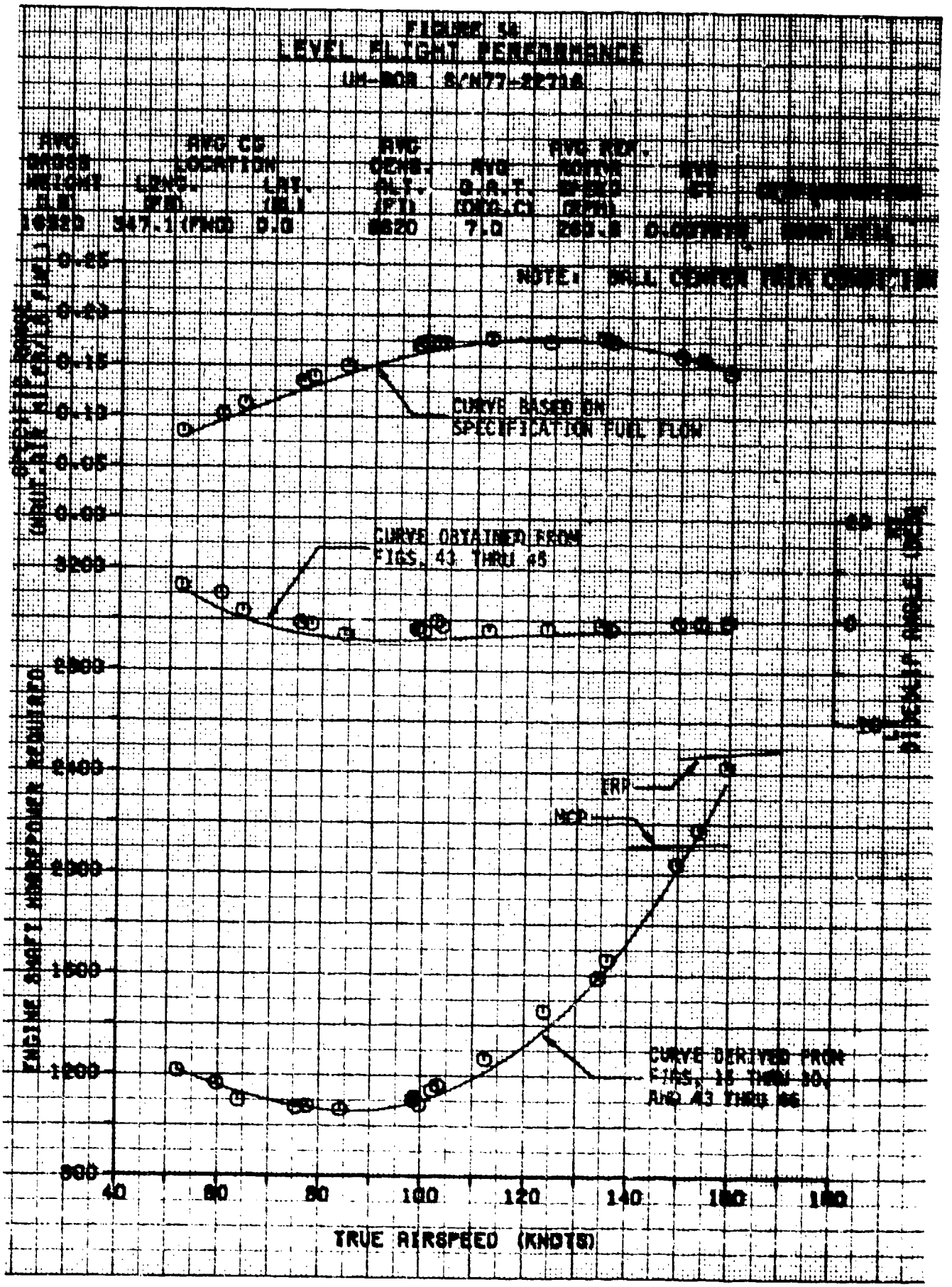


FIGURE 54
LEVEL FLIGHT PERFORMANCE
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
ERP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30 AND 43 THRU 66
TRUE AIRSPEED (KNOTS)

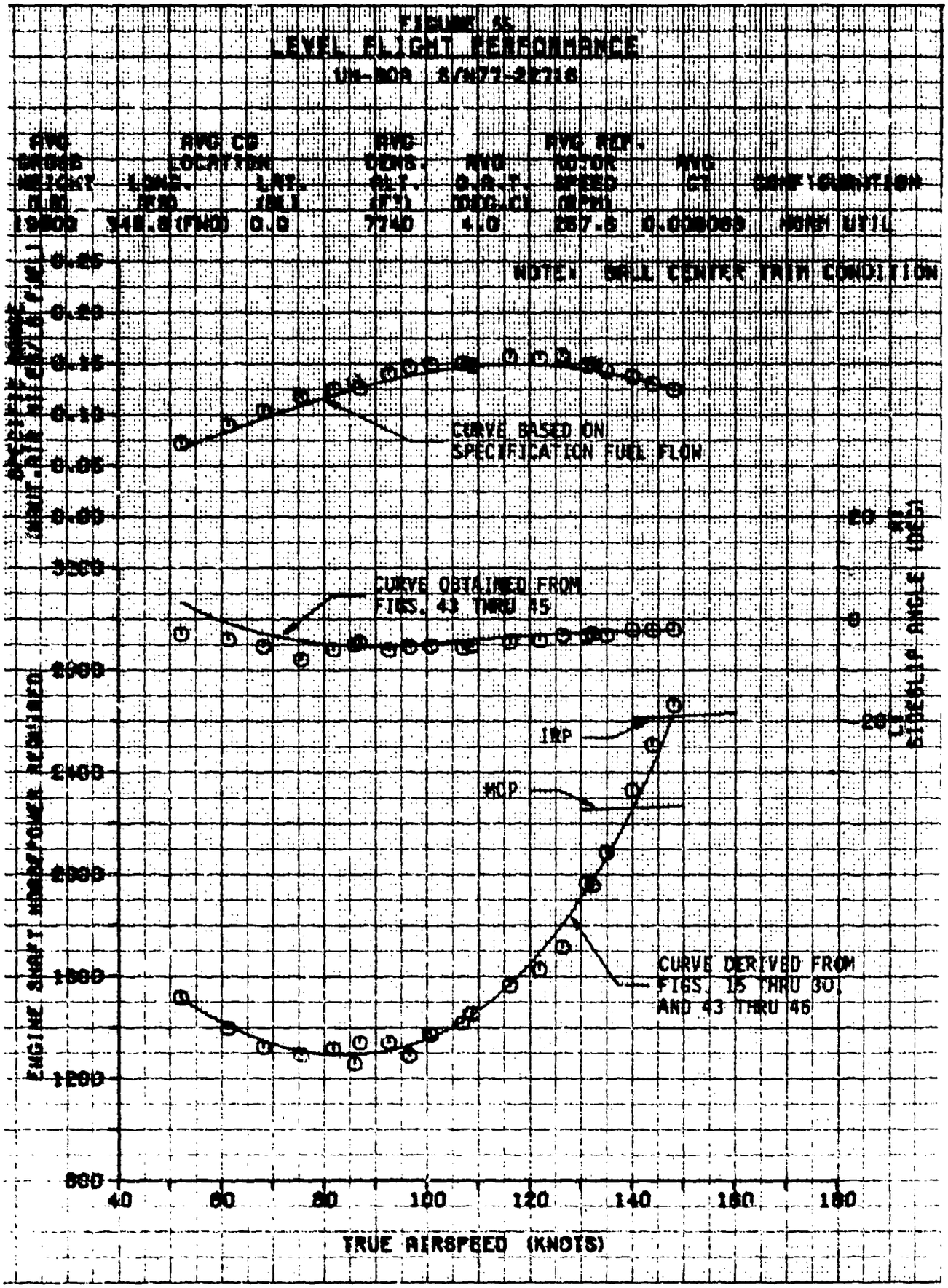


FIGURE 46
LEVEL FLIGHT PERFORMANCE
UH-60A S/N77-22716
NOTE: DRIL CENTER TRIM CONDITION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
ENGINE SHAFT HORSEPOWER REQUIRED
SIDESLIP ANGLE (DEG)
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180

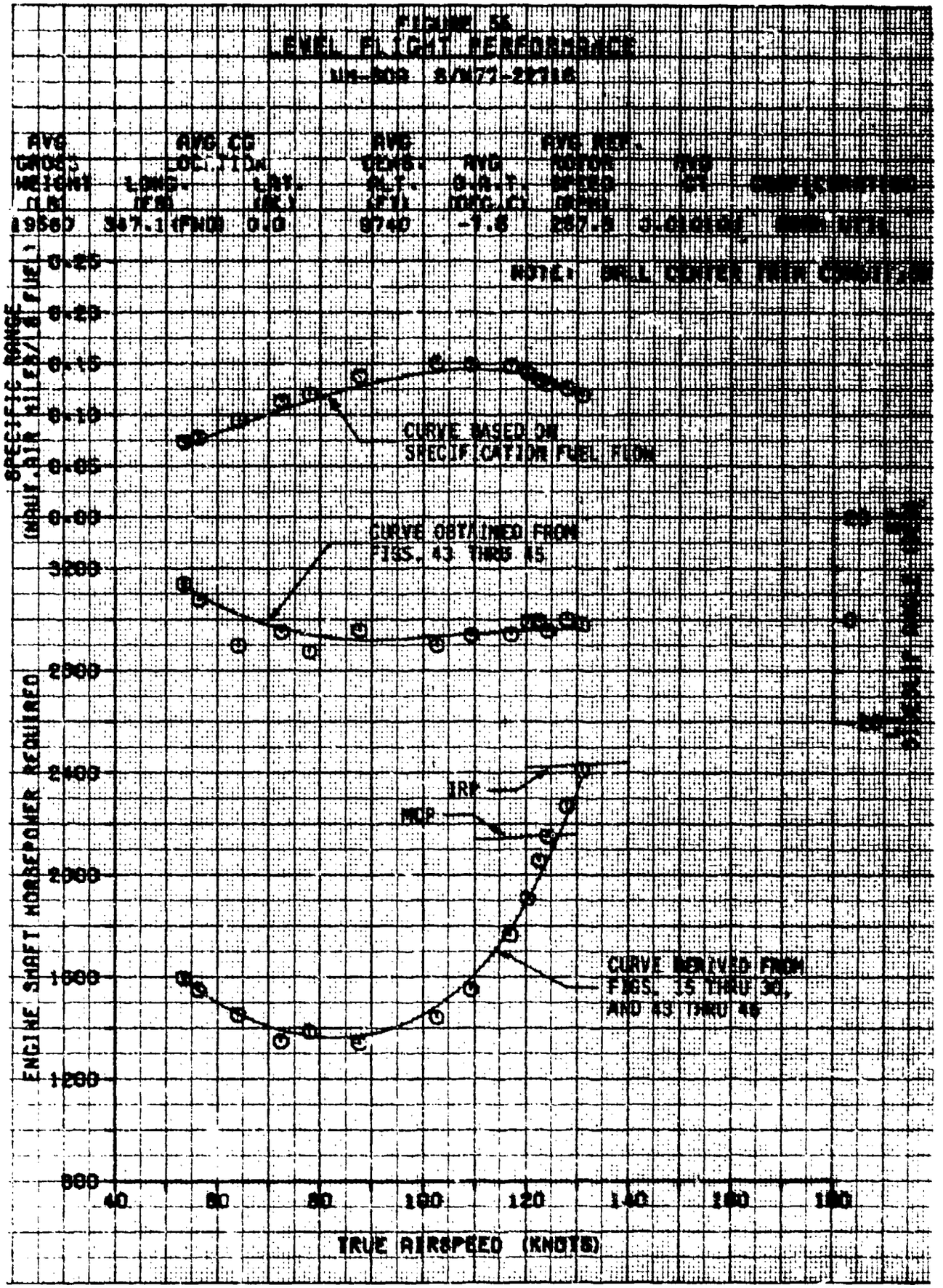
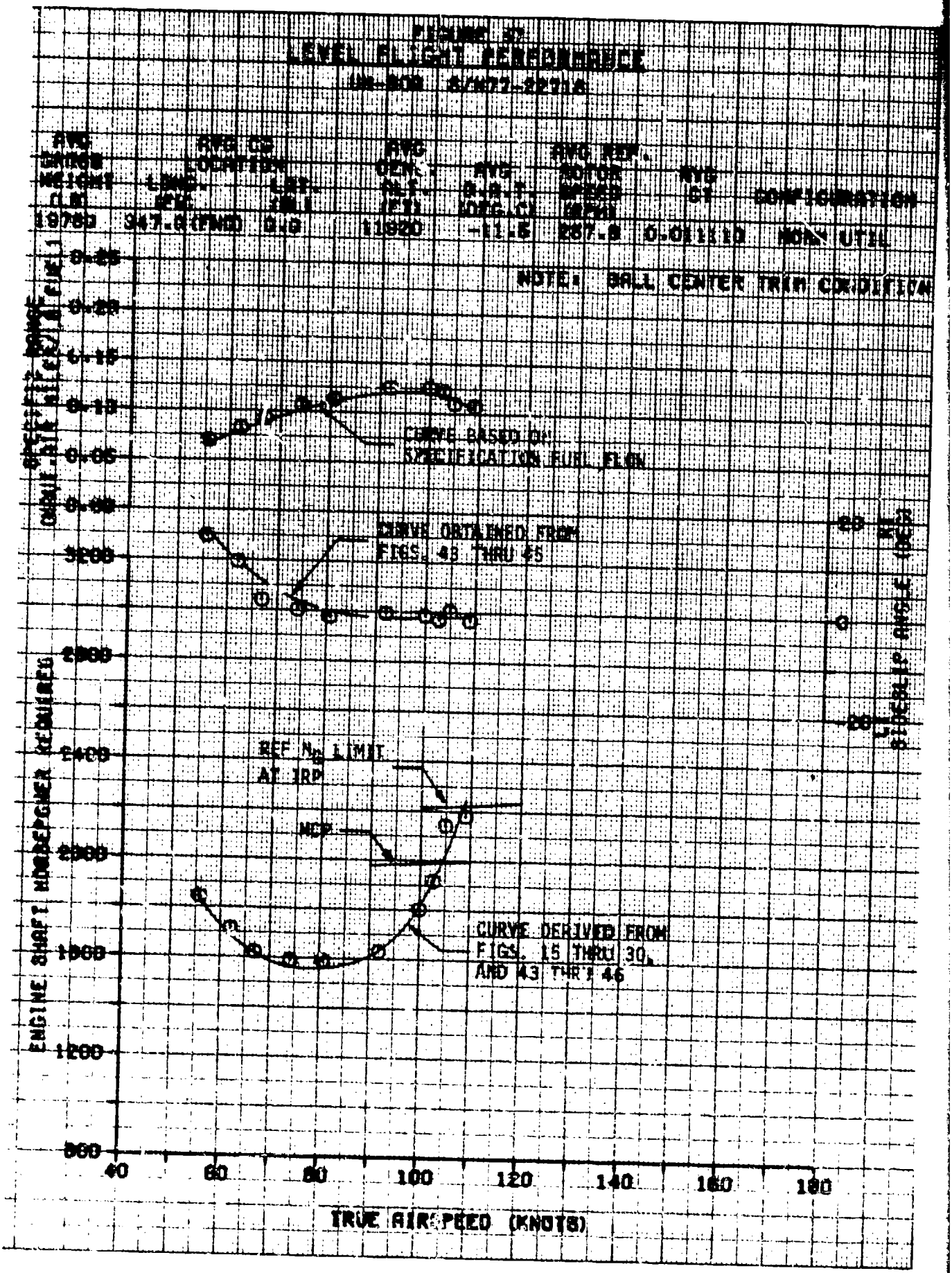


FIGURE 56.
FUEL FLIGHT PERFORMANCE
AVG GROSS WEIGHT (LBS)
AVG CG LOCATION LONG. (FS)
LAT. (BL)
AVG GROSS WT. (LBS)
AVG N.P. (FS)
AVG N.A.T. (DEG.C)
AVG REF. ROTOR SPEED (RPM)
9560
347.1
FWD
0.0
9740
-7.6
257.8
NOTE: DRILL CENTER FWD CONFIGURATION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 45
SPECIFIC RANGE (NAUT AIR MILES/LB FUEL)
ENGINE SHAFT HORSEPOWER REQUIRED
3000
2500
2000
1500
1000
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)


LEVEL FLIGHT PERFORMANCE
AVG GROSS WEIGHT LBS.
AVG CG LOCATION LONG. STA. IN.
AVG DENS. ALT. (FT)
AVG R.O.T. DEG/SEC
AVG REF. ROTOR SPEED (RPM)
AVG GT
CONFIGURATION
19780 347.6 (FWD) 0.0 11920 -11.5 257.8 0.0111 10 NORM UTIL
NOTE: BALL CENTER TRIM CONDITION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 48 THRU 65
REF N_R LIMIT AT IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
ENGINE SHAFT HORSEPOWER REQUIRED
CHANGE IN FUEL FLOW (LB/HR)
CHANGE IN ATTITUDE ANGLE (DEG)
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180

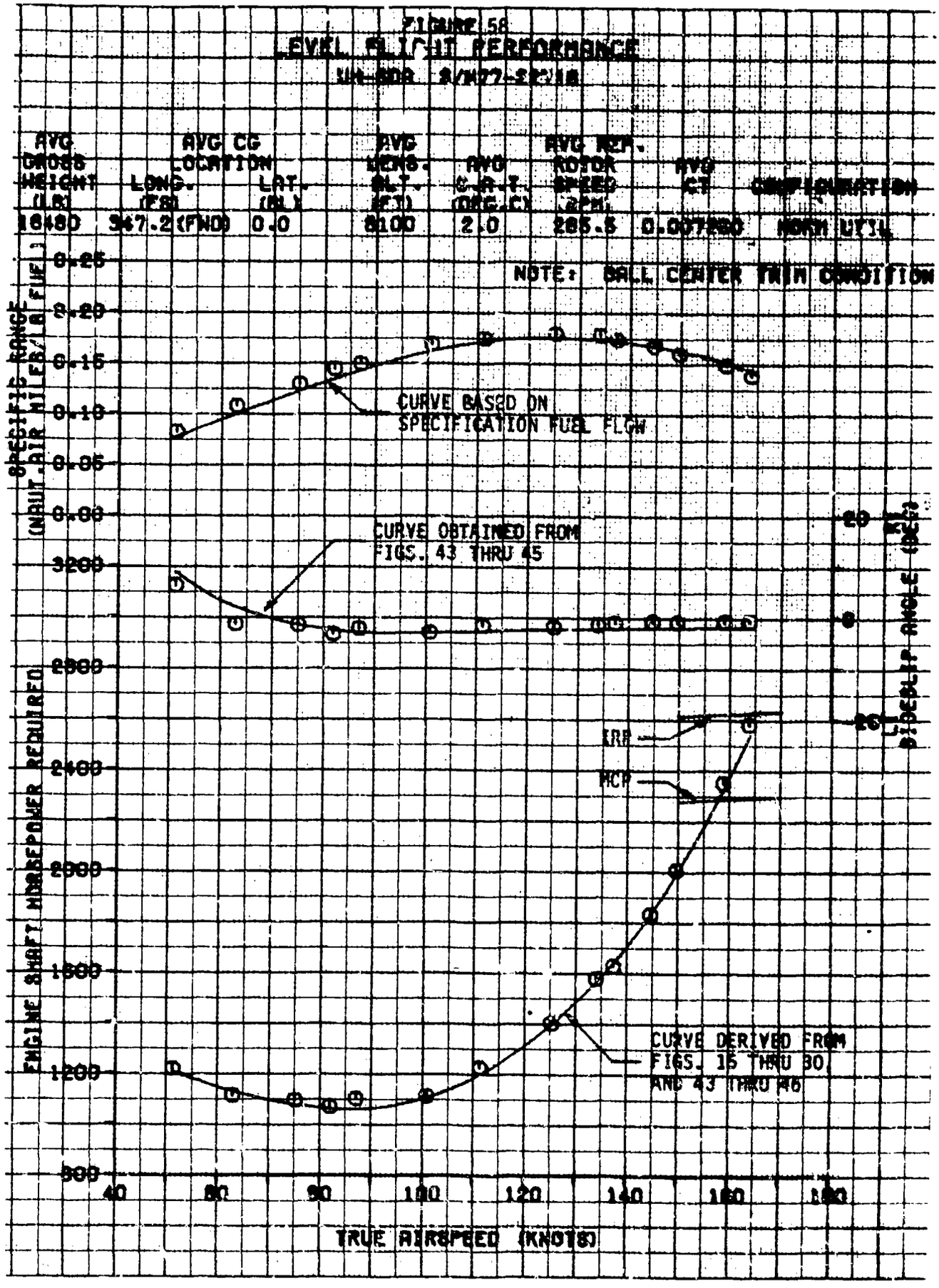


FIGURE 58
LEVEL FLIGHT PERFORMANCE
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG. (FS)
LAT. (BL)
AVG ALT. (FT)
AVG C.A.T. (DEG.C)
AVG REF. ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
16480 347.2 (FWD) 0.0 8100 2.0 285.5 0.007280 NORM UTIL
NOTE: BALL CENTER TRIM CONDITION
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
400
IRP
MCP
BIDEGLIP ANGLE (DEG)
20
0
-20
CURVE DERIVED FROM FIGS. 15 THRU 30 AND 43 THRU 46
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

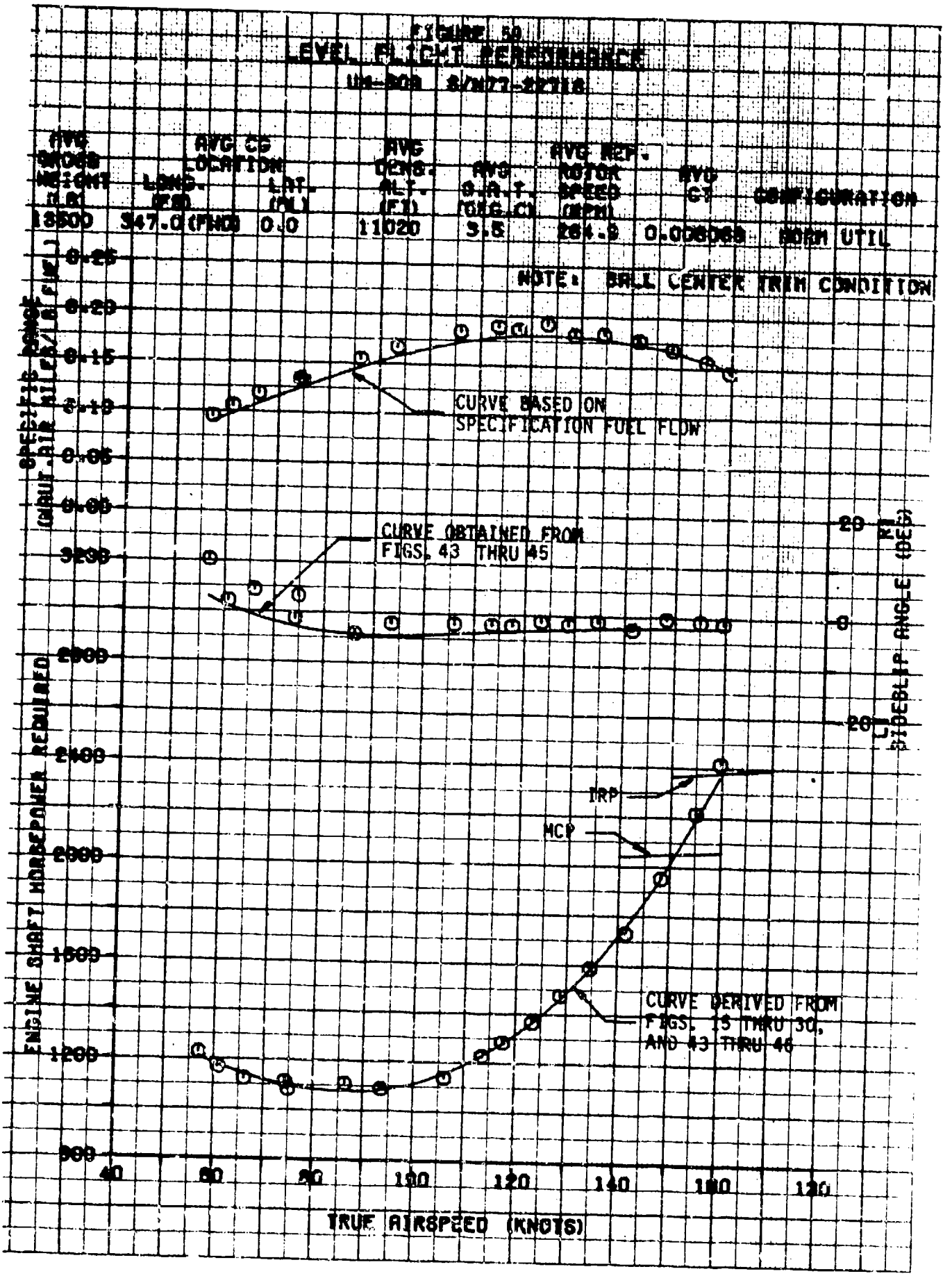


FIGURE 50
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22718
AVG GROSS WEIGHT (LB) 13500
AVG CG LOCATION LONG. (FS) 347.0 (FWD)
LAT. (BL) 0.0
AVG DENS. ALT. (FT) 11020
AVG O.A.T. (DEG.C) 3.5
AVG ROTOR SPEED (RPM) 264.9
AVG GT 0.000000
CONFIGURATION NORM UTIL
NOTE: DRILL CENTER TRIM CONDITION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
ENGINE SHAFT HORSEPOWER REQUIRED
TIP/BLIP ANGLE (DEG)
TRUE AIRSPEED (KNOTS)

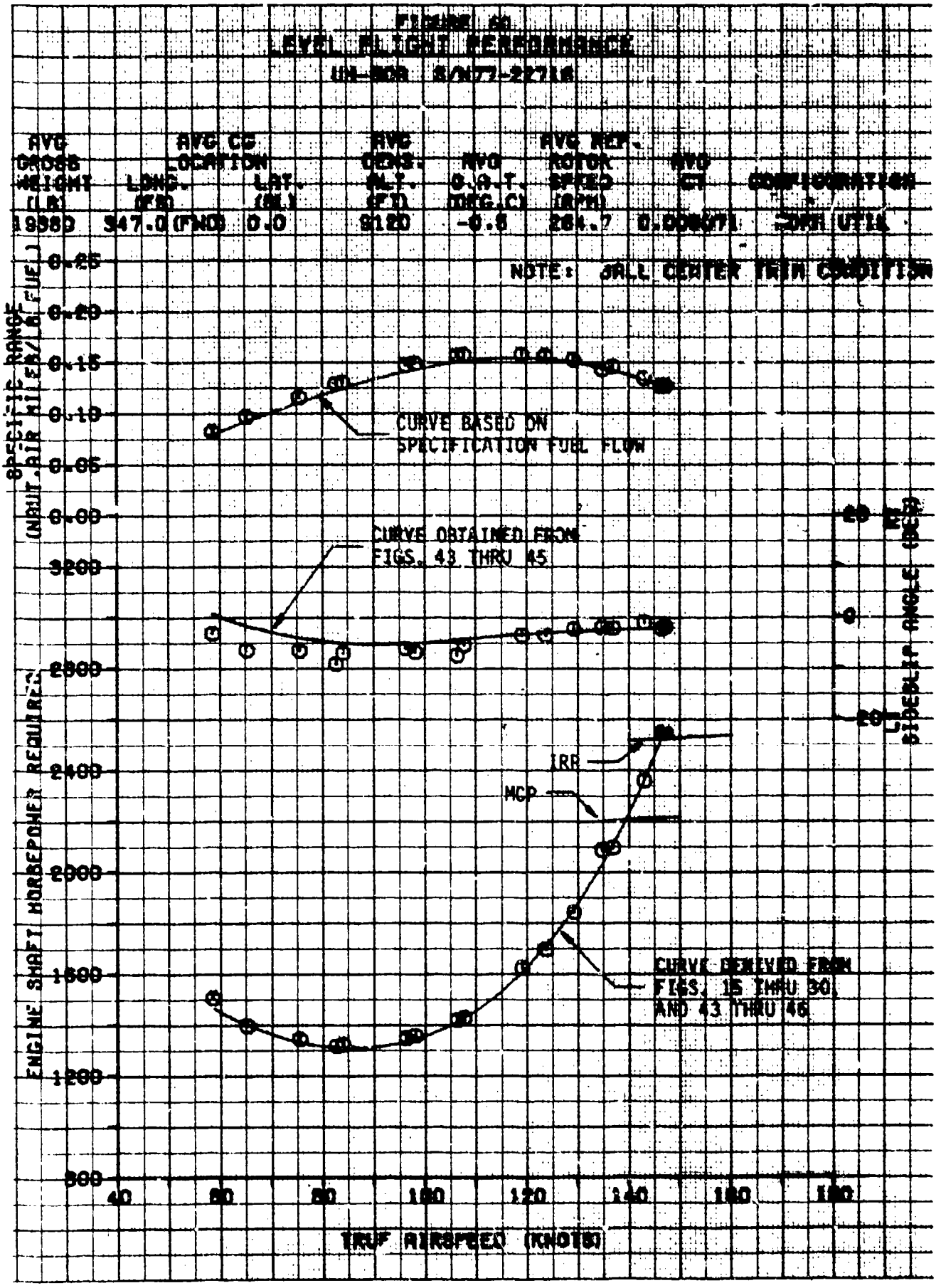


FIGURE 48
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22716
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG. (FS)
LAT. (BL)
AVG DENS. ALT. (FT)
AVG A.S.T. DEG.C)
AVG ROTOR SPEED (RPM)
AVG GT
CONFIGURATION
19380 347.0(FWD) 0.0 6120 -0.8 264.7 0.000071 .20MM UTIL
NOTE: ALL CENTER TRIM CONDITION
SPECIFIC RANGE (NAUT.AIR MILES/LB FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

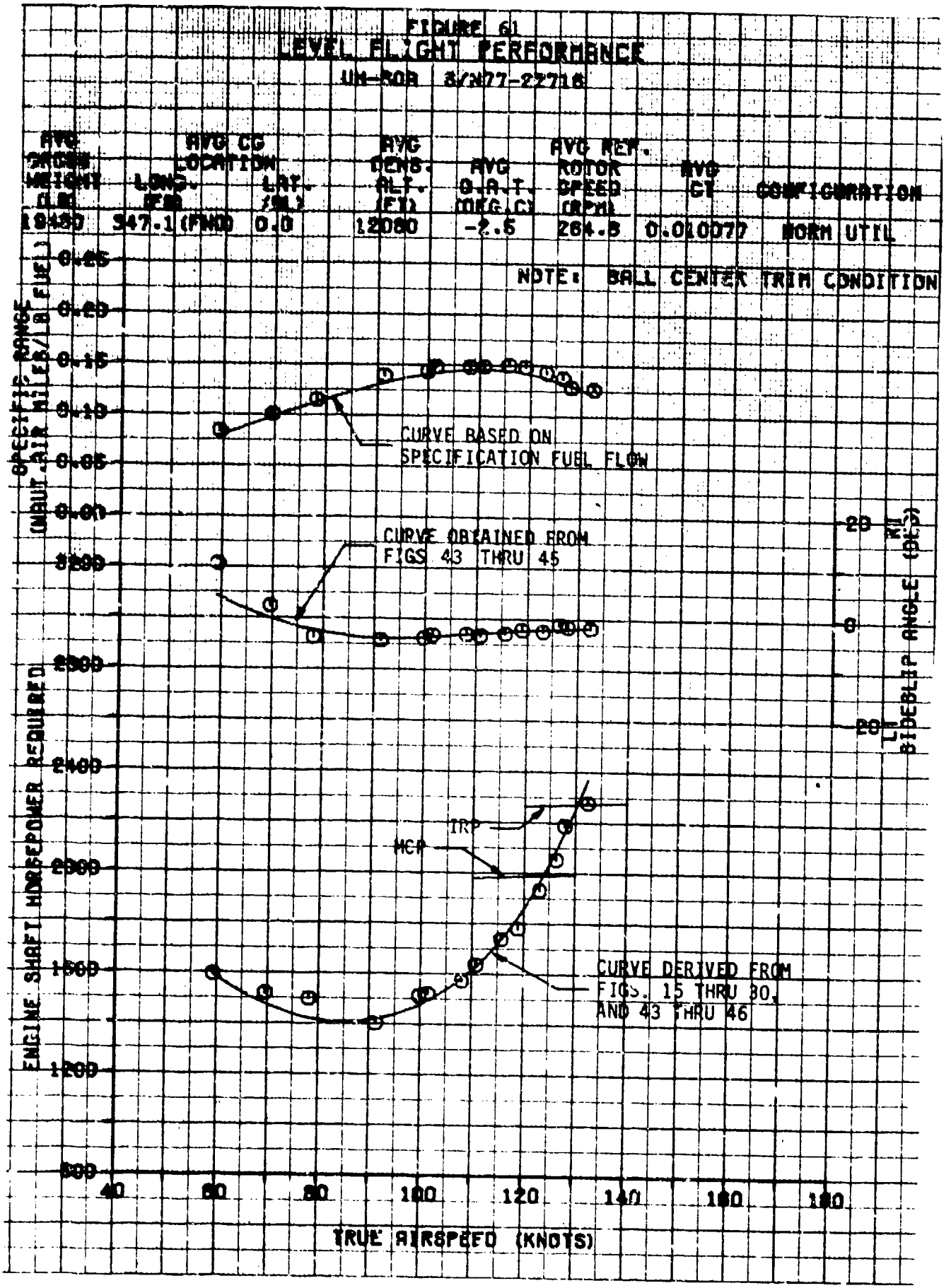
FIGURE 61

LEVEL FLIGHT PERFORMANCE

UH-60A S/N77-22716

AVG GROSS WEIGHT (LBS)	AVG CG LOCATION LONG. (FS)	AVG CG LOCATION LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG GT	CONFIGURATION
19450	347.1 (FWD	0.0	12080	-2.5	264.8	0.010077	NORM UTIL

NOTE: BALL CENTER TRIM CONDITION



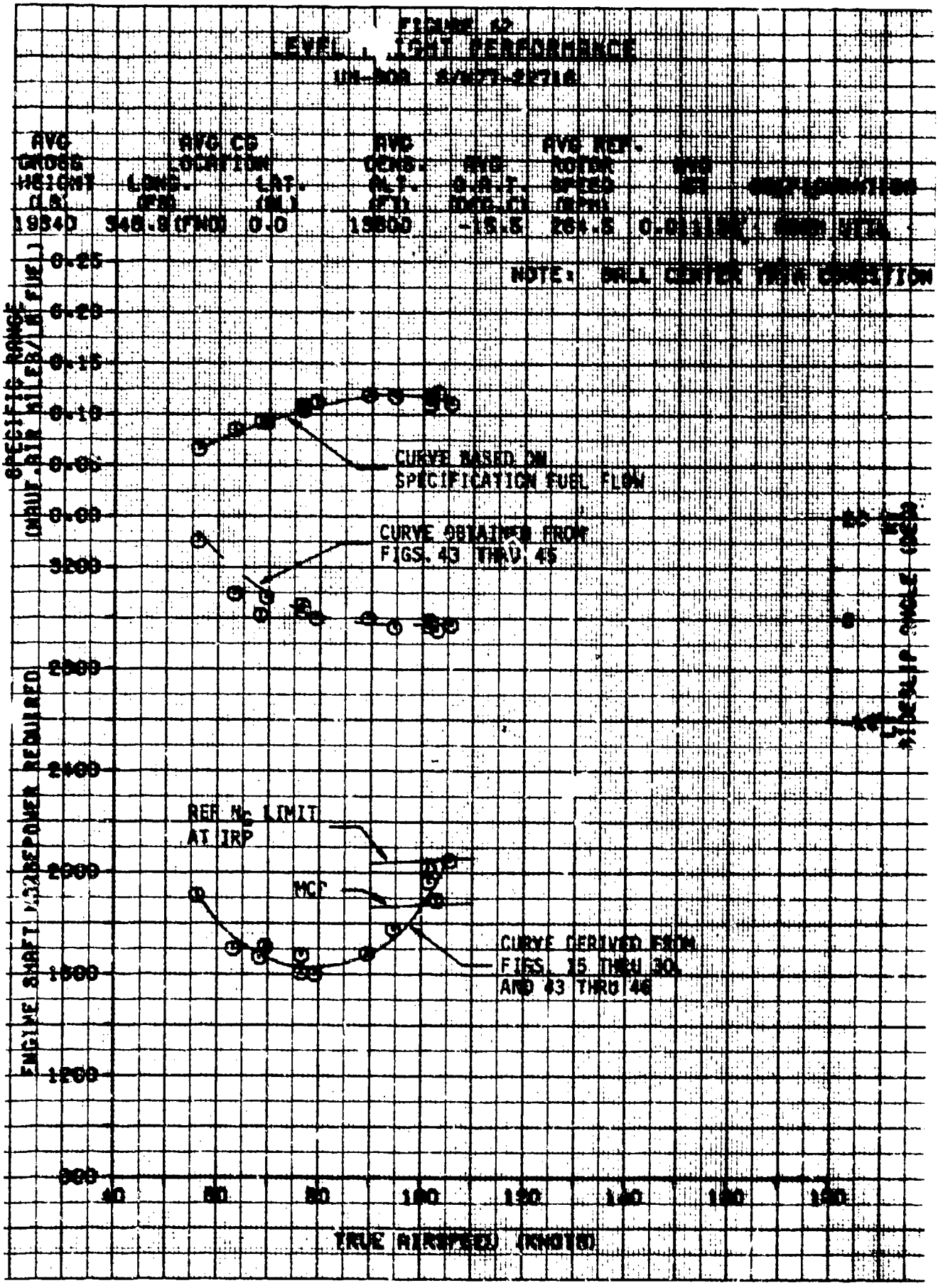


FIGURE 42
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22718
AVG GROSS WEIGHT (LBS)
AVG CG LOCATION LONG. (FS)
LAT. (BL)
AVG DENS. ALT. (FT)
AVG O.A.T. (DEG.C)
AVG ROT. SPEED (RPM)
19340
348.8 (FWD)
0.0
15800
-15.5
264.5
NOTE: ROLL CENTER TRIM CONDITION
SPECIFIC FUEL FLOW (POUNDS PER HOUR PER POUND OF FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
3200
2800
2400
2000
1600
1200
800
ENGINE SHAFT HORSEPOWER REQUIRED
REF N_c LIMIT AT IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30 AND 43 THRU 46
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

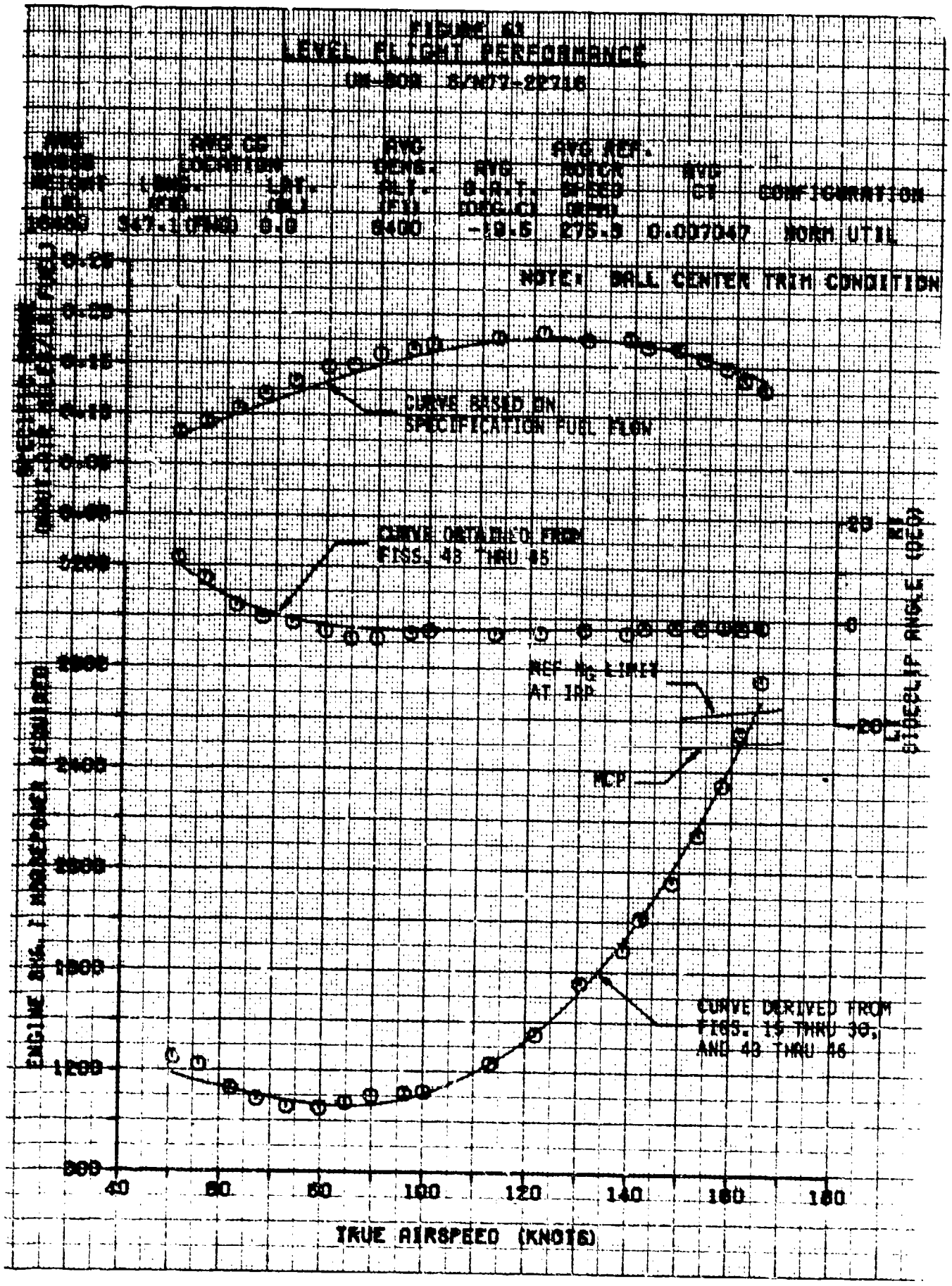


FIGURE 44
LEVEL FLIGHT PERFORMANCE
NOTE: BALL CENTER TRIM CONDITION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 85
REF Nc LIMIT AT IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
ENGINE SHAFT HORSEPOWER (EACH ENGINE)
FLIGHT PATH ANGLE (DEG)
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180
1200
1800
2400
-20
0
20

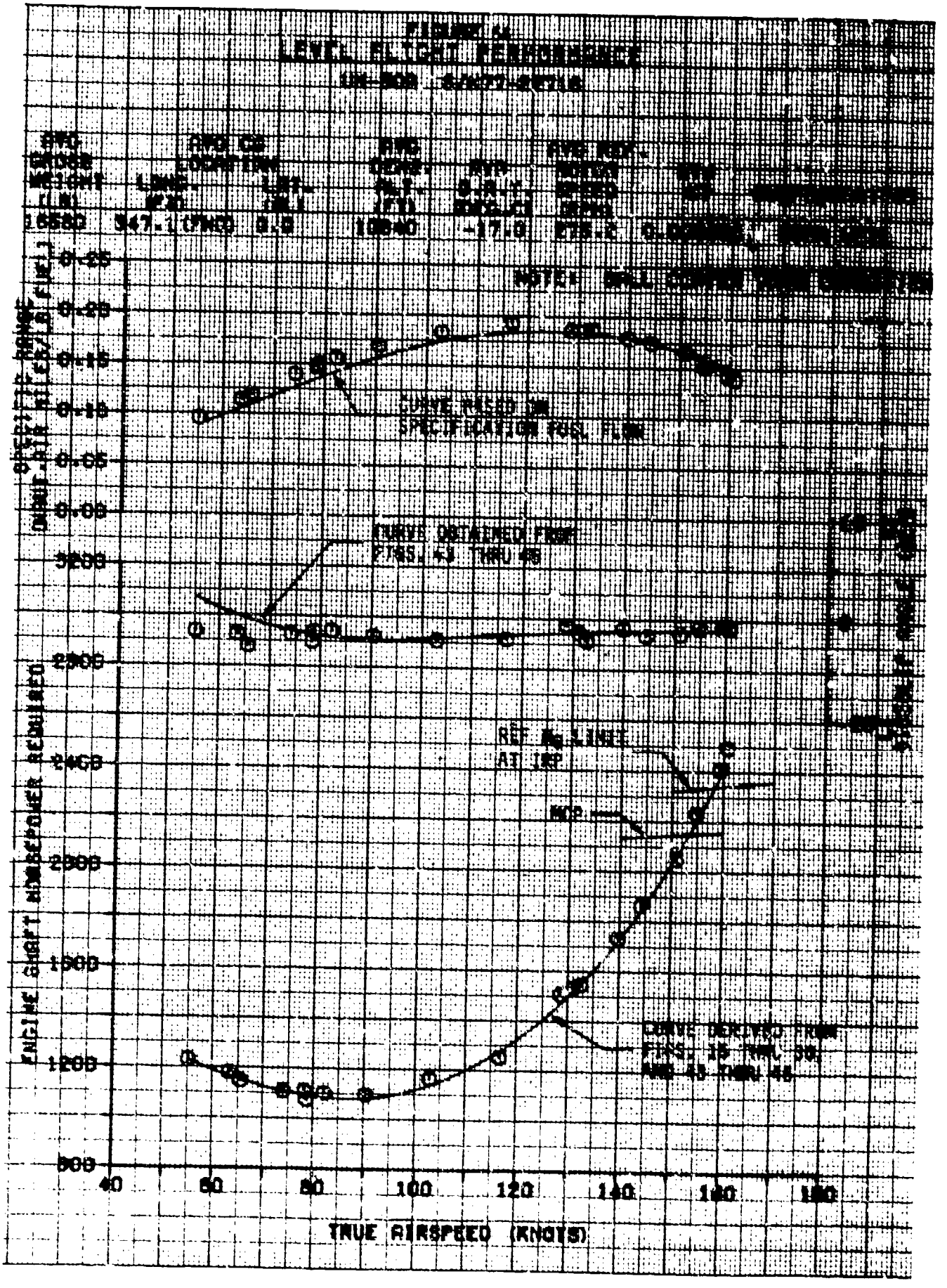


FIGURE 44
LEVEL FLIGHT PERFORMANCE
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 46
REF Pa LINE AT IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 39, AND 43 THRU 46
ENGINE SHAFT HORSEPOWER REQUIRED
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180
500
1000
1500
2000
2500
3000

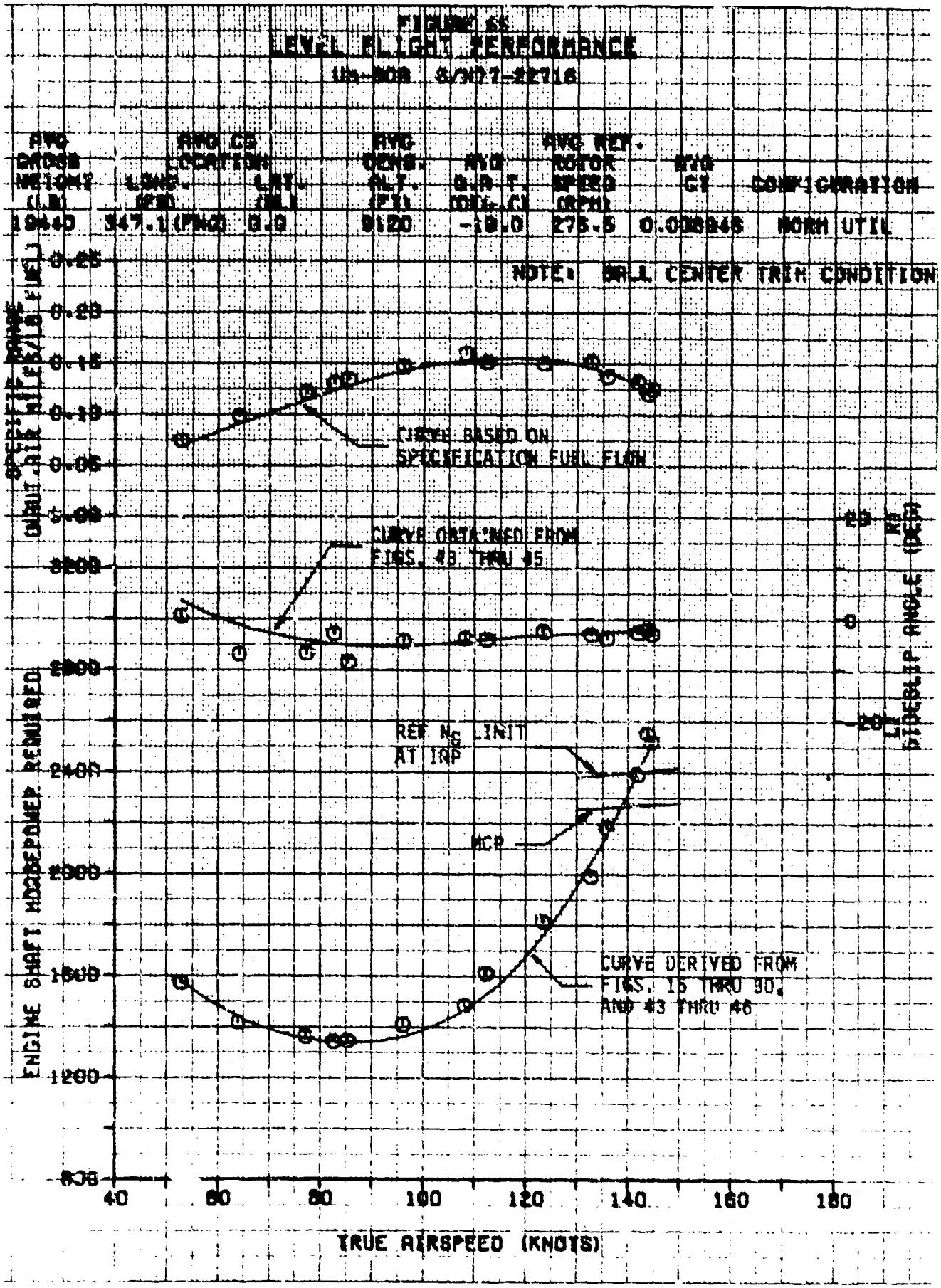


FIGURE 65
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22716
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG. (IN. FWD)
LAT. (IN.)
AVG DENS. ALT. (FT)
AVG O.A.T. DEV. (°C)
AVG ROTOR SPEED (RPM)
AVG Ct
CONFIGURATION
16440
347.1 (FWD)
0.0
9120
-10.0
275.5
0.006945
NORM UTIL
NOTE: BRL CENTER TRIM CONDITION
SPECIFIC FUEL CONSUMPTION (POUNDS PER HOUR PER SHAFT HORSEPOWER)
ENGINE SHAFT HORSEPOWER REQUIRED
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
REF N_R LIMIT AT IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
REF ROTOR TIP-PATH-PLANE ANGLE (DEG)
0.25
0.20
0.15
0.10
0.05
3200
2800
2400
2000
1600
1200
800
20
0
-20
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

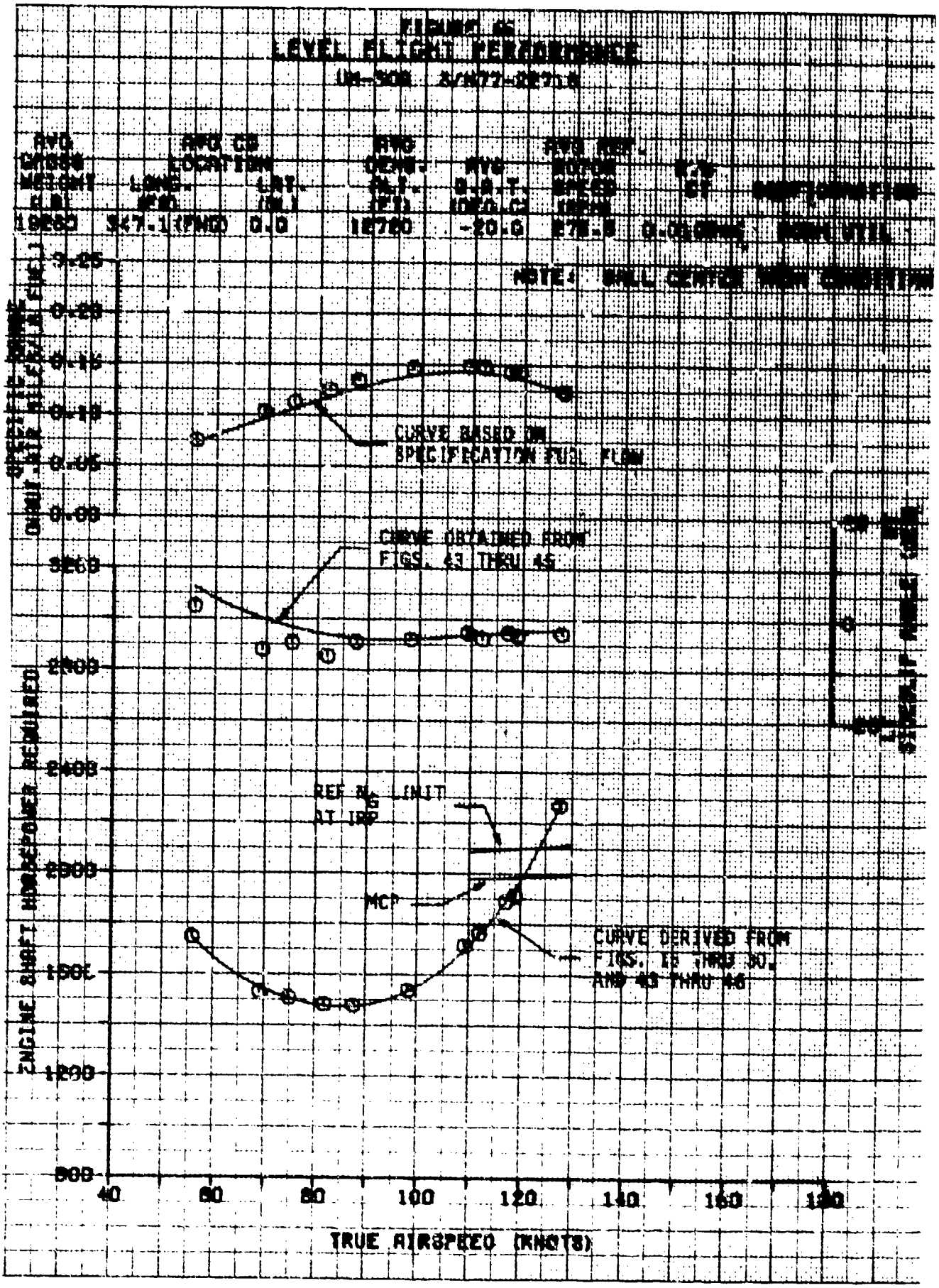


FIGURE 46
LEVEL FLIGHT PERFORMANCE
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
REF n_z LIMIT AT IRP
MCP
CURVE DERIVED FROM FIGS. 16 THRU 30, AND 40 THRU 46
ENGINE SHAFT HORSEPOWER REQUIRED
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160

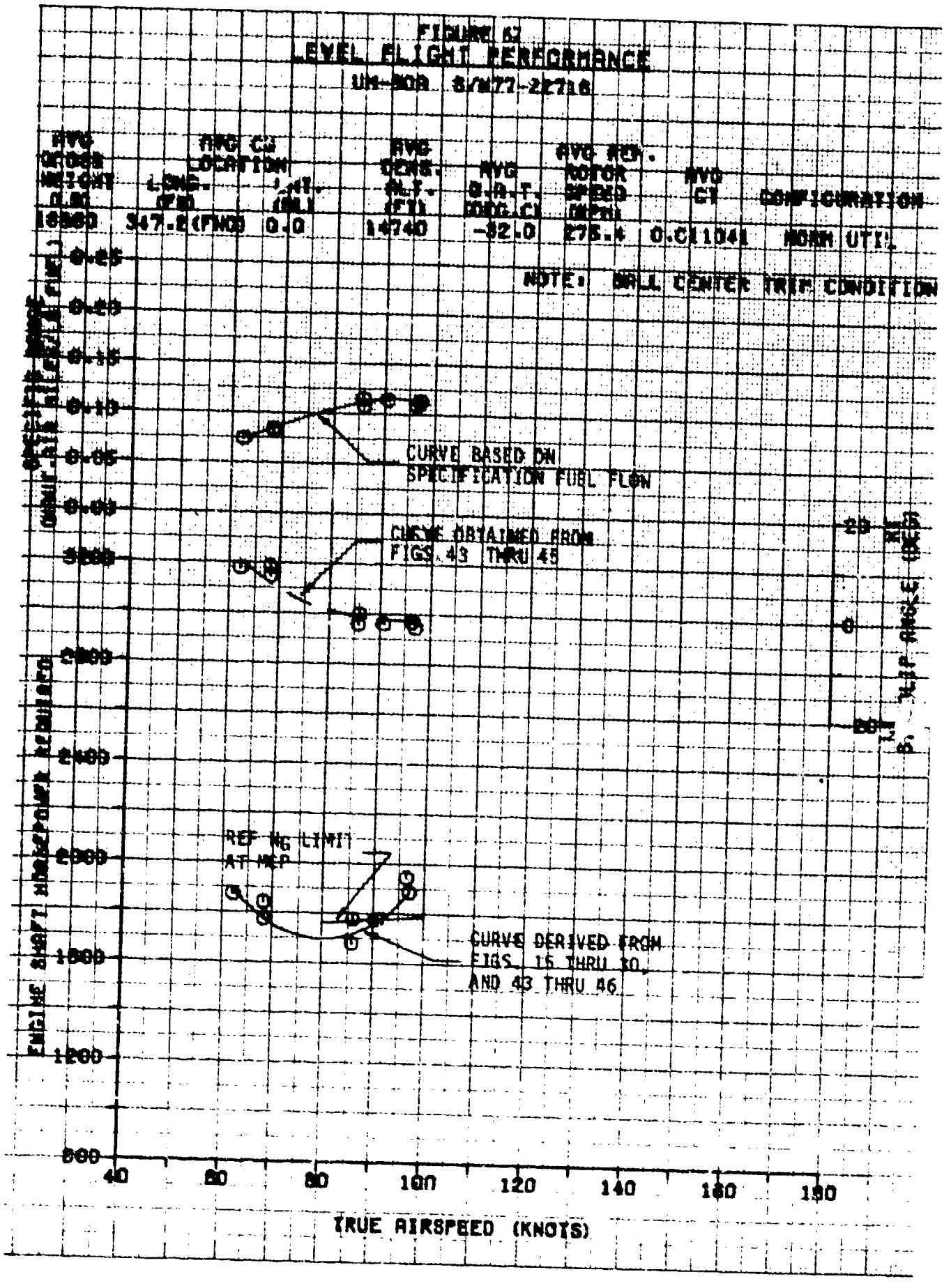


FIGURE 62
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22716
AVG GROSS WEIGHT (LBS) 16560
AVG CG LOCATION LONG. (FS) 347.8(FWD)
LAT. (BL) 0.0
AVG DENS. ALT. (FT) 14740
AVG O.A.T. (DEG.C) -32.0
AVG ROT. SPEED (RPM) 275.4
AVG FOR. CG 0.CG1041
CONFIGURATION NORM UTIL
NOTE: DRILL CENTER TRIP CONDITION
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
REF NG LIMIT AT MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND 43 THRU 46
ENGINE SHAFT HORSEPOWER REQUIRED
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180
2000
1800
1200
800

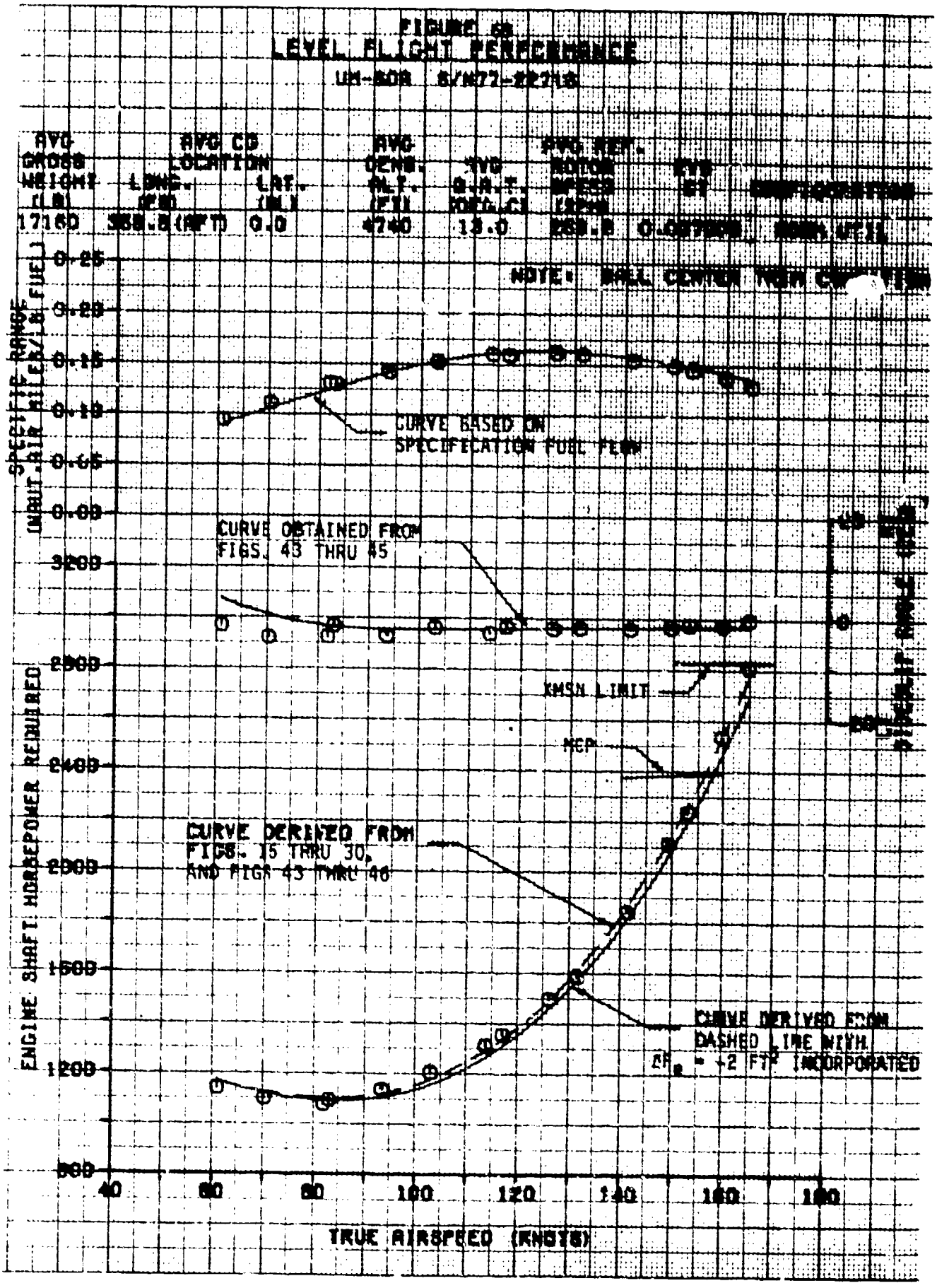


FIGURE 68
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22716
AVG GROSS WEIGHT (LB) 17160
AVG CG LOCATION LONG. (FS) 368.8 (AFT)
LAT. (BL) 0.0
AVG DENS. ALT. (FT) 4740
AVG G.A.T. (DEG.C) 13.0
AVG ROTOR SPEED (RPM) 258.8
NOTE: BALL CENTER FROM CURVE
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
3200
2800
2400
2000
1600
1200
800
KMSN LIMIT
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND FIGS. 43 THRU 46
ENGINE SHAFT HORSEPOWER REQUIRED
CURVE DERIVED FROM DASHED LINE WITH ΔH_p = +2 FT² INCORPORATED
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

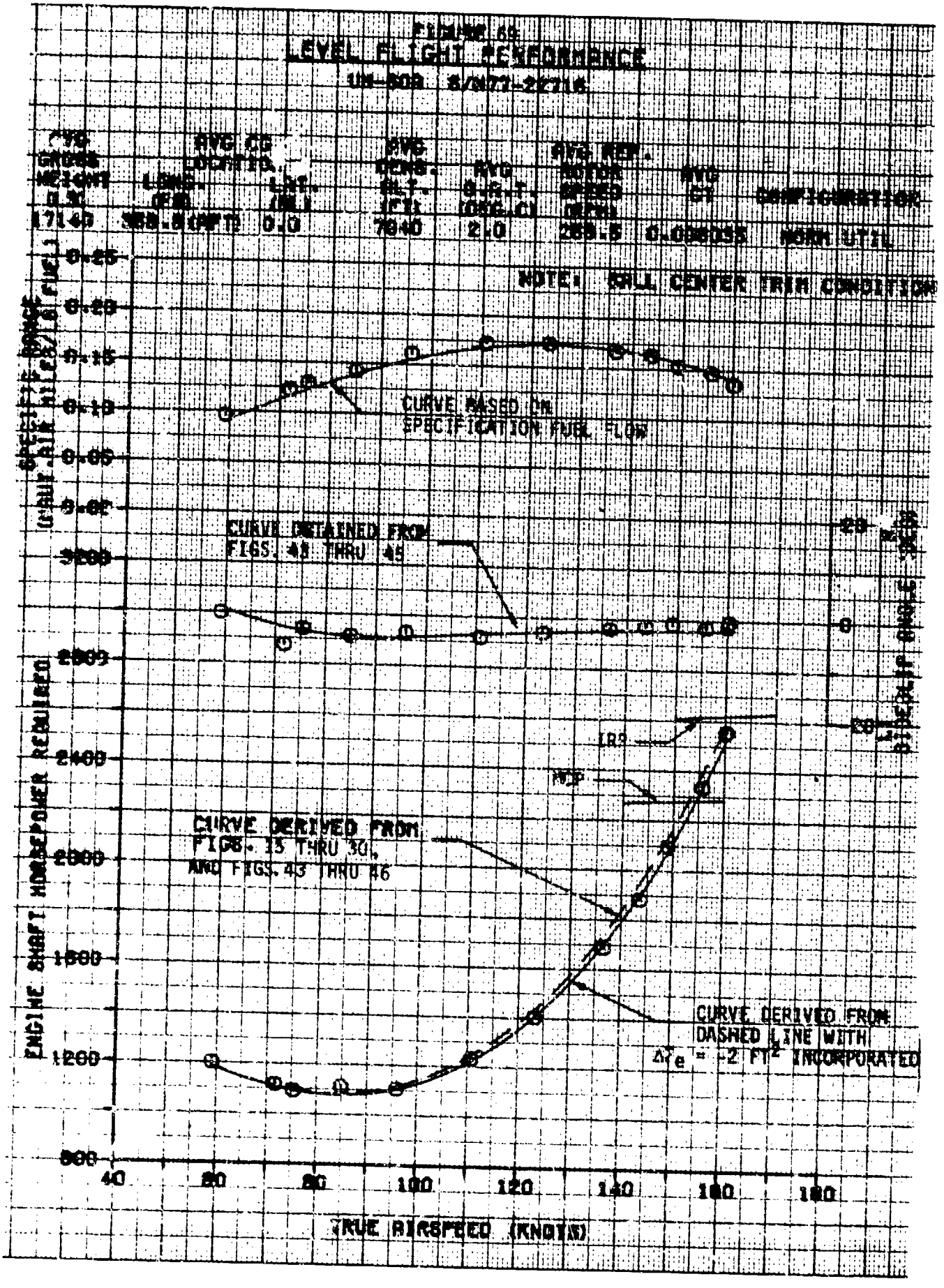


FIGURE 46
LEVEL FLIGHT PERFORMANCE
UH-60A S/N 77-22716
AVG GW
CROSS
WEIGHT
17140
AVG CG
LONG.
STA.
358.6 INCHES
LAT.
BL
0.0
AVG
ALT.
(FT)
7840
AVG
R.A.T.
(DEG.C)
2.0
AVG ROTOR
SPEED
(RPM)
258.5
AVG
GT
0.000655
CONFIGURATION
NORM UTIL
NOTE: CGLL CENTER TRIM CONDITION
CURVE BASED ON
SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM
FIGS. 43 THRU 45
CURVE DERIVED FROM
FIGS. 15 THRU 30,
AND FIGS. 43 THRU 46
LRC
MCP
CURVE DERIVED FROM
DASHED LINE WITH
Δf_e = +2 FT^2 INCORPORATED
SPECIFIC RANGE (NAUT MILES/LB OF FUEL)
0.25
0.20
0.15
0.10
0.05
0
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

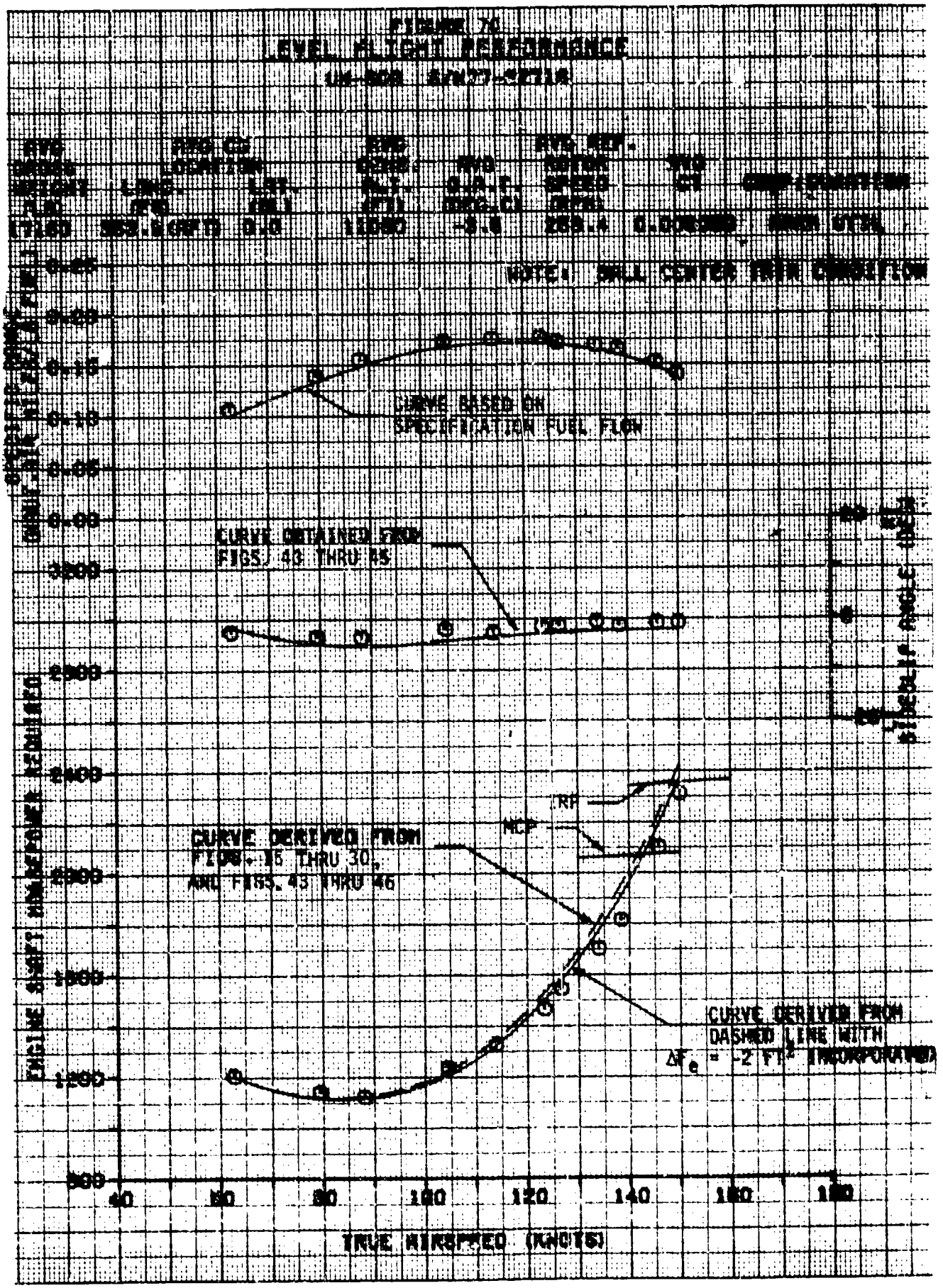


FIGURE 70
LEVEL FLIGHT PERFORMANCE
NOTE: SKILL CENTER RPM CONDITION
CURVE BASED ON
SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM
FIGS. 43 THRU 45
CURVE DERIVED FROM
FIGS. 15 THRU 30,
AND FIGS. 43 THRU 46
TRF
MCP
CURVE DERIVED FROM
DASHED LINE WITH
ΔF_e = -2 FT^2 INCORPORATED
ENGINE SHAFT HORSEPOWER REQUIRED
TRUE AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180

FIGURE 31

LEVEL FLIGHT PERFORMANCE

US-60A BuNo 22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
16540	347.2 (FWD)	0.0	5960	11.5	258.5	0.007047	DOORS OPEN

NOTE: BALL CENTER TRIM CONDITION

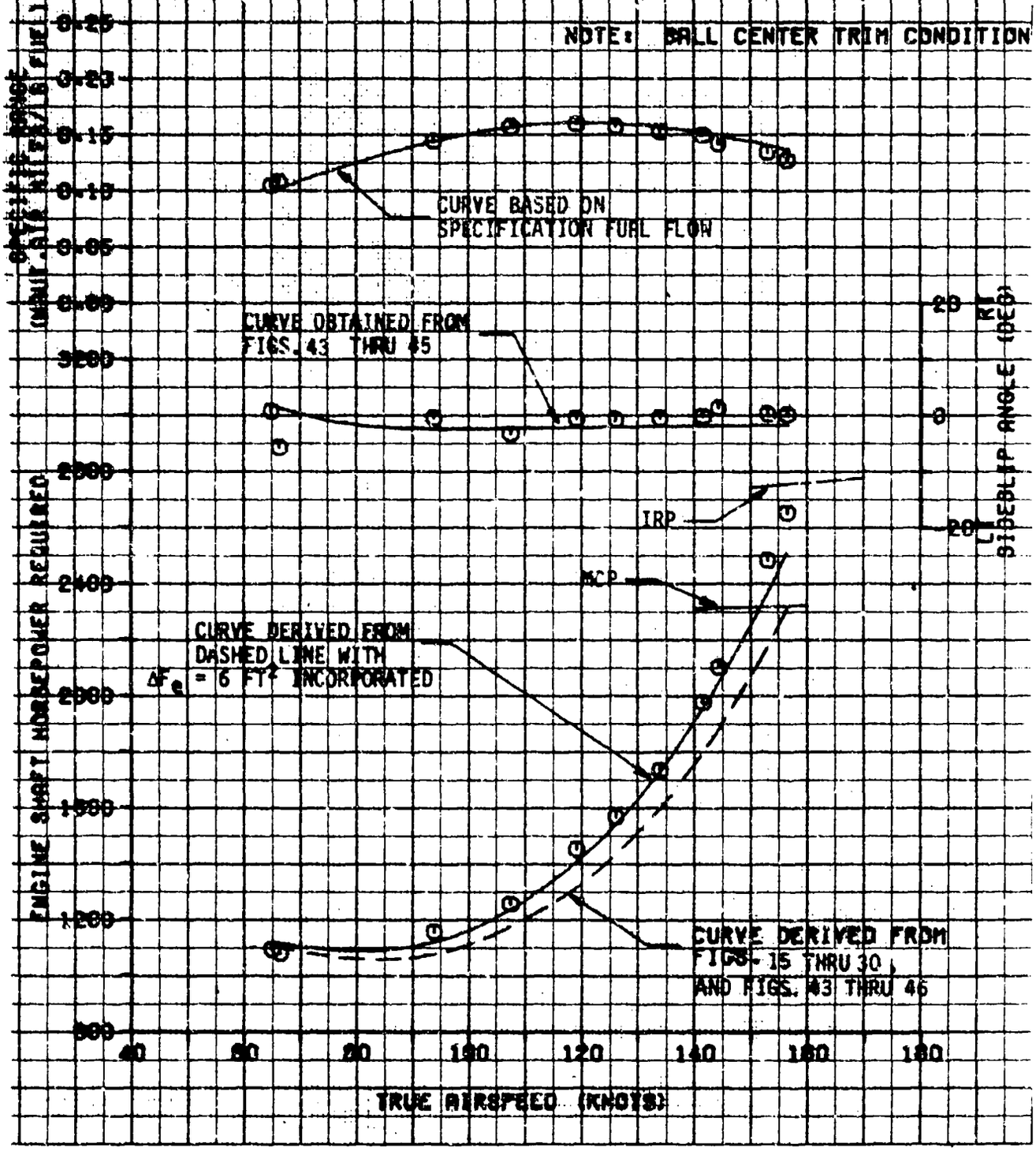


FIGURE 22

LEVEL FLIGHT PERFORMANCE

AH-60A S/N77-22718

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG.C)	AVG REF. ROTOR SPEED (RPM)	AVG GW	CONFIGURATION
16360	347.1 (FWD)	0.0	3020	-10.6	258.9	0.0036050	DOORS OPEN

NOTE: BALL CENTER TRIM CONDITION

SPECIFIC RANGE (NAUT AIR MILES/LB FUEL)

0.25
0.20
0.15
0.10
0.05
0.00

CURVE BASED ON SPECIFICATION FUEL FLOW

CURVE OBTAINED FROM FIGS 43 THRU 45

REF N_G LIMIT AT IRP

MCP

CURVE DERIVED FROM DASHED LINE WITH ΔF_e = 6 FT^2 INCORPORATED

CURVE DERIVED FROM FIGS. 15 THRU 30, AND FIGS. 43 THRU 45

ENGINE SHAFT HORSEPOWER REQUIRED

3200
2800
2400
2000
1600
1200
800

40 60 80 100 120 140 160 180

TRUE AIRSPEED (KNOTS)

FIGURE 73

LEVEL FLIGHT PERFORMANCE

UH-60A S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
16480	347.3 (FWD)	0.0	12320	-1.0	258.4	0.009058	DOORS OPEN

NOTE: BALL CENTER TRIM CONDITION

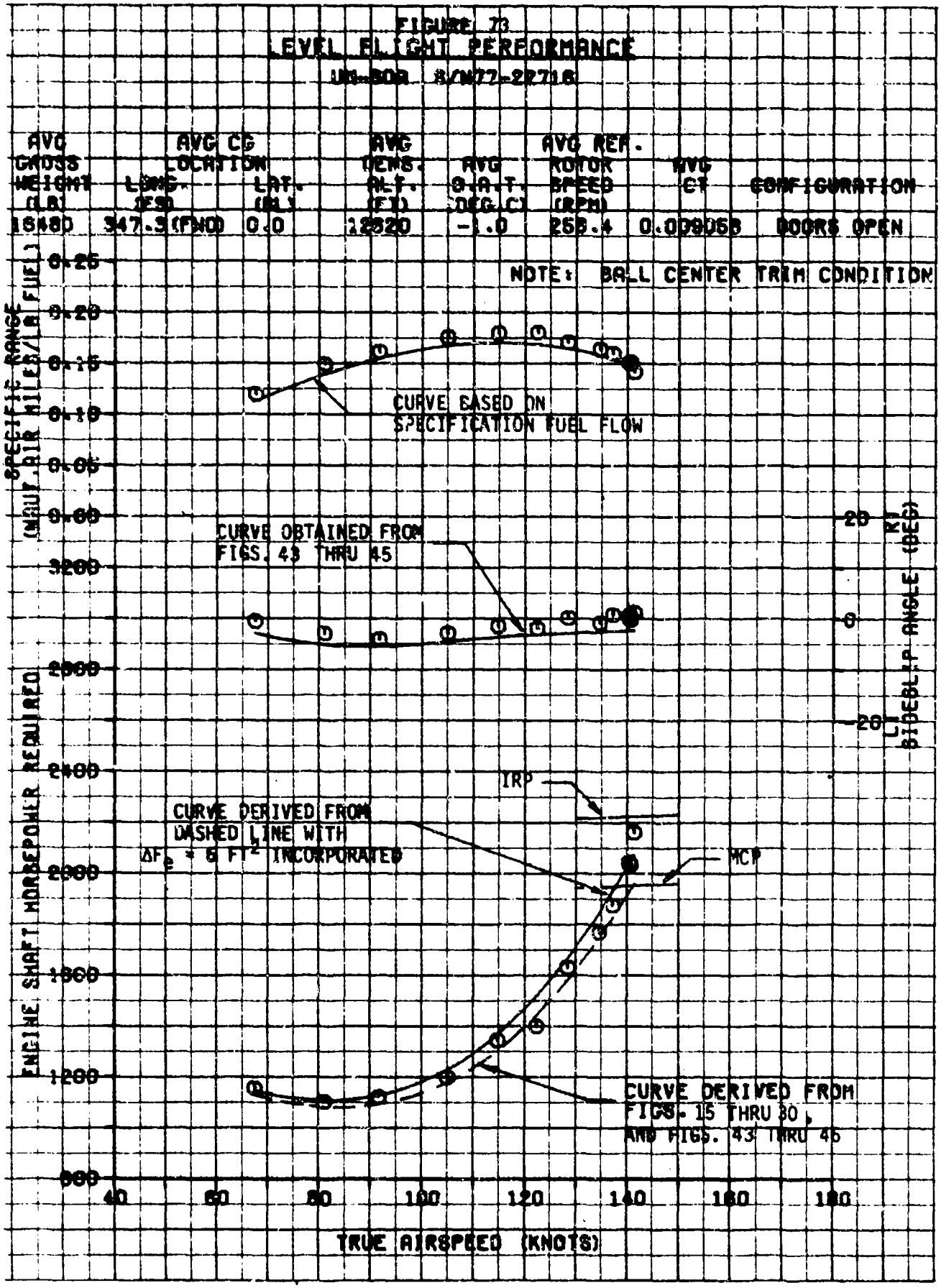


FIGURE 74

EVEL FLIGHT PERFORMANCE

UH-60A S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG C)	AVG REF. ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
18880	347.2 (FWD)	0.0	9160	9.5	258.2	0.008020	IR

NOTE: BALL CENTER TRIM CONDITION

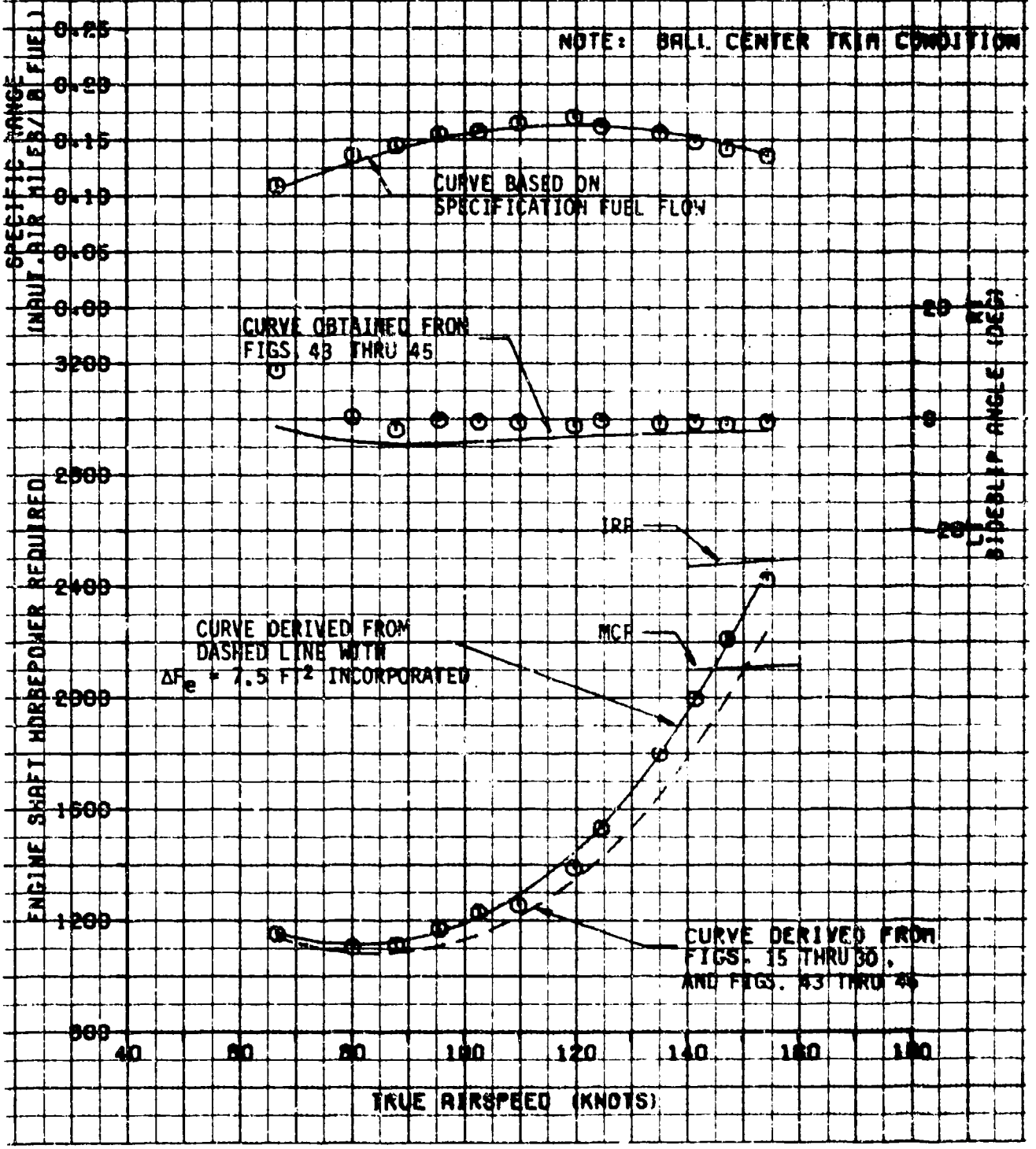


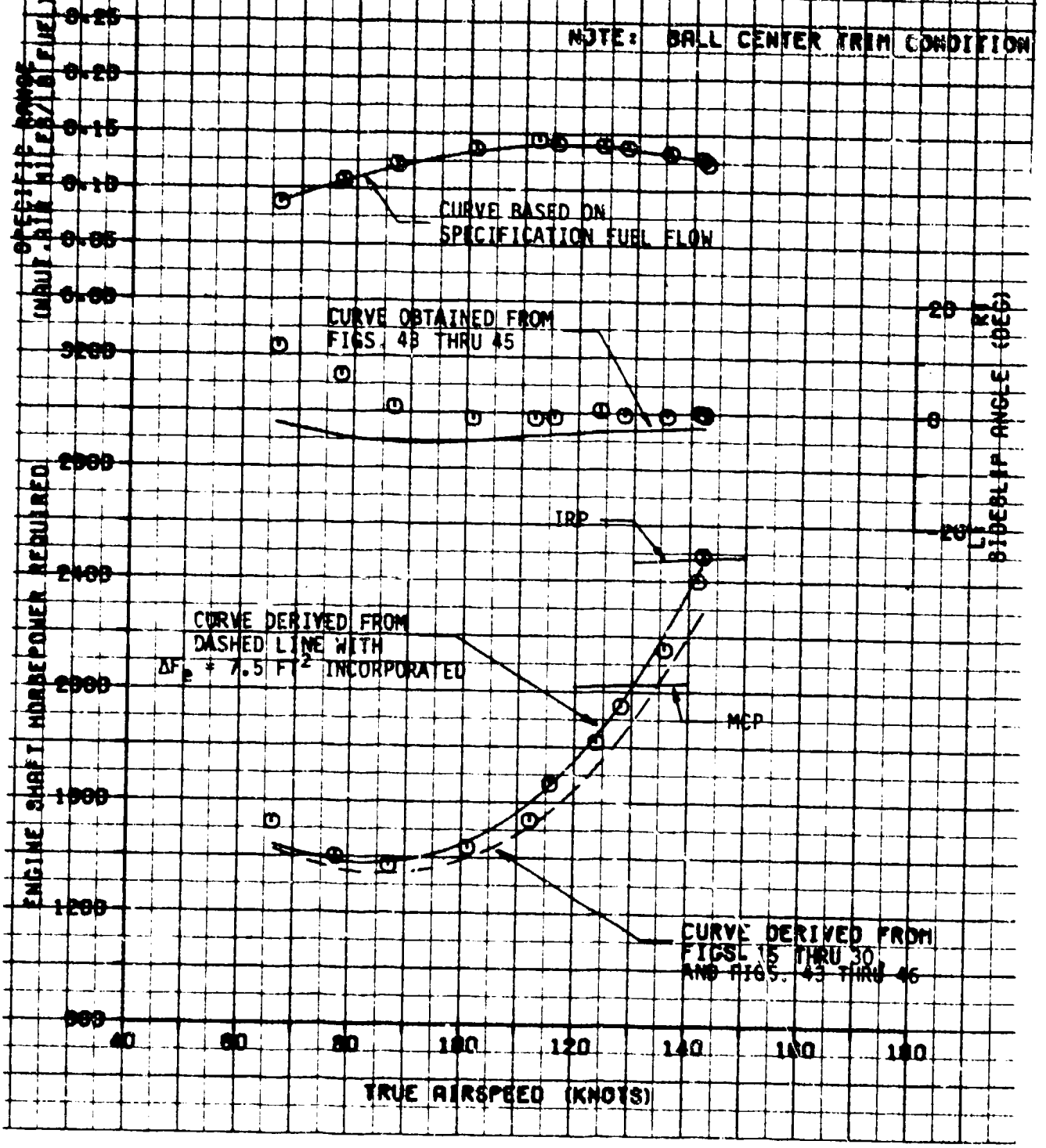
FIGURE 75

LEVEL FLIGHT PERFORMANCE

UH-60A S/N 77-22718

AVG GROSS WEIGHT (LBS)	AVG CG LOCATION LONG. (FS)	LAT. (BL)	AVG DENS. ALT. (FT)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG GT	CONFIGURATION
19800	347.1 (FWD)	0.0	6780	17.6	258.2	0.009040	IR

NOTE: BALL CENTER TRIM CONDITION



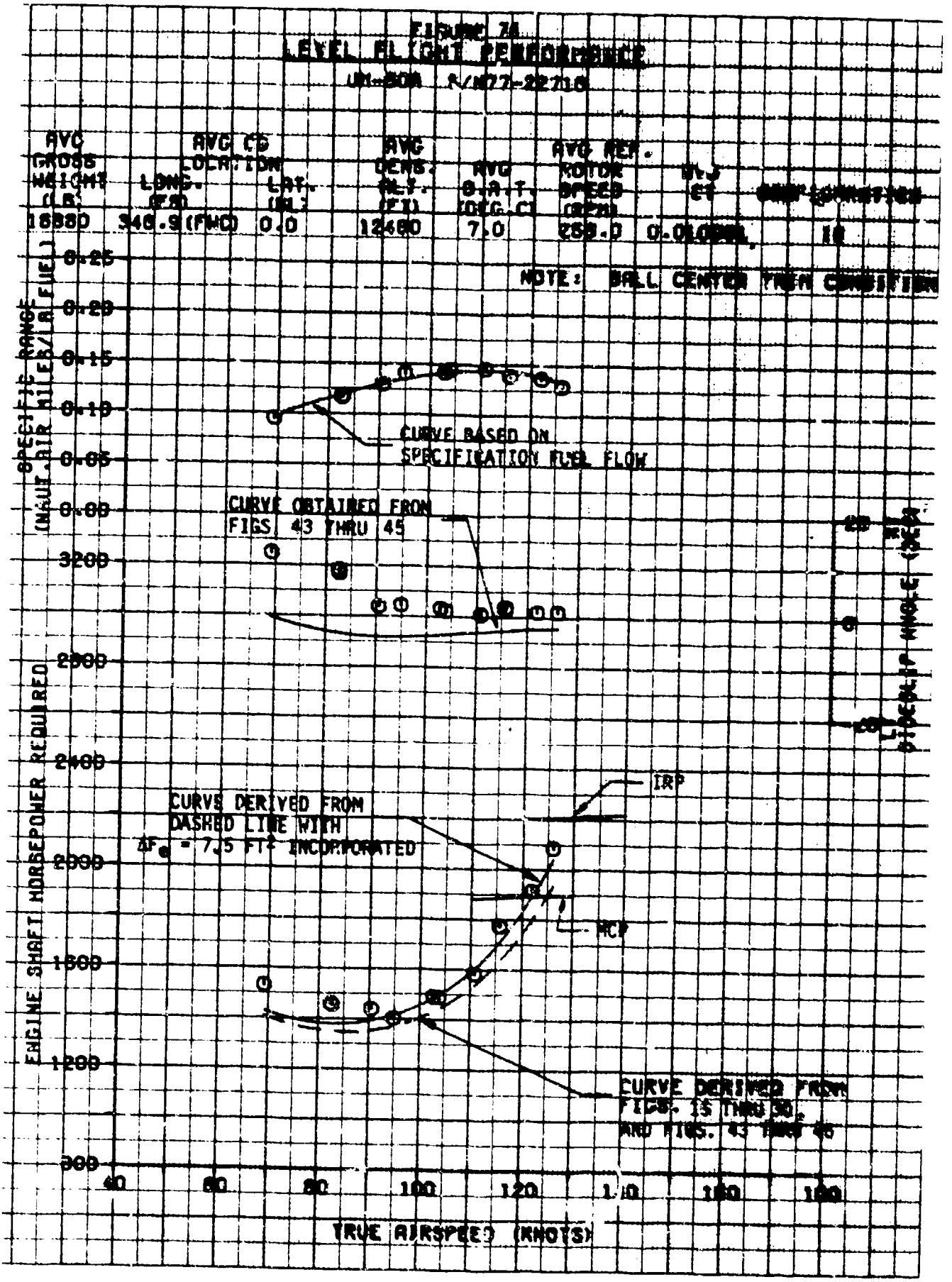


FIGURE 7A
LEVEL FLIGHT PERFORMANCE
UH-60A S/N77-22710
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG. (FS)
LAT. (BL)
AVG DENS. ALT. (FT)
AVG O.A.T. (DEG.C)
AVG REF. ROTOR SPEED (RPM)
16880
346.9 (FWD)
0.0
12480
7.0
258.0
NOTE: BALL CENTER YAW CONDITION
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
CURVE BASED ON SPECIFICATION FUEL FLOW
CURVE OBTAINED FROM FIGS. 43 THRU 45
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
CURVE DERIVED FROM DASHED LINE WITH ΔF_e = 7.5 FT² INCORPORATED
IRP
MCP
CURVE DERIVED FROM FIGS. 15 THRU 30, AND FIGS. 43 THRU 45
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)

FIGURE 77

AUTOROTATIONAL DESCENT PERFORMANCE

UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T
⊙	16,200	348.4(FWD)	-0.2(LT)	5520	6.0	256	0.006865
⊡	19,680	347.3(FWD)	-0.2(LT)	5120	8.0	254	0.008369

NOTE: NORMAL UTILITY CONFIGURATION

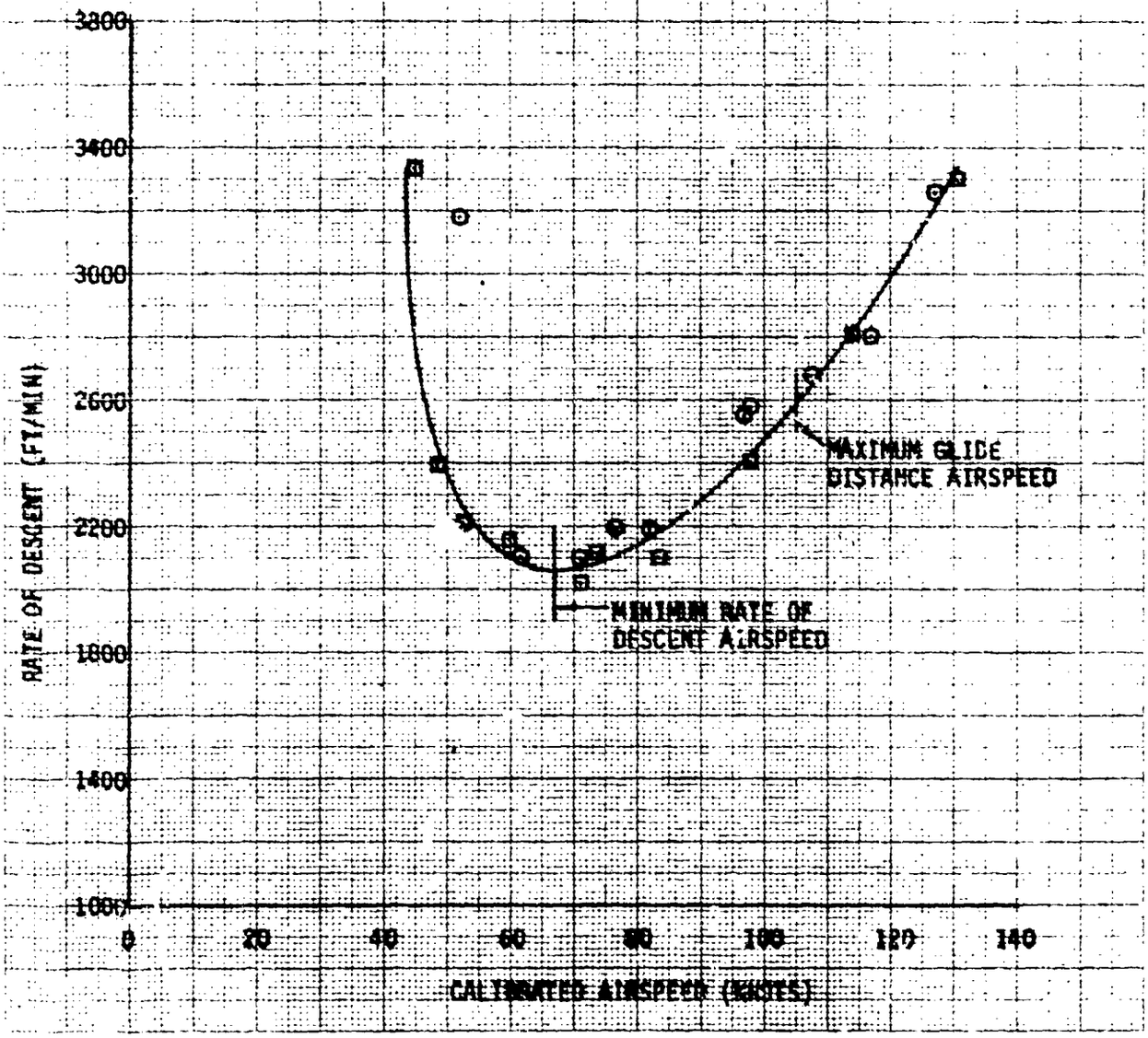


FIGURE 78

AUTOROTATIONAL DESCENT PERFORMANCE

UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG H_D (FT)	AVG OAT (°C)	AVG CALIBRATED AIRSPEED (KT)
○	16,500	347.5(FWD)	-0.2(LT)	5460	4.5	76
□	13,700	347.4(FWD)	-0.2(LT)	5460	8.5	73

NOTE: NORMAL UTILITY CONFIGURATION

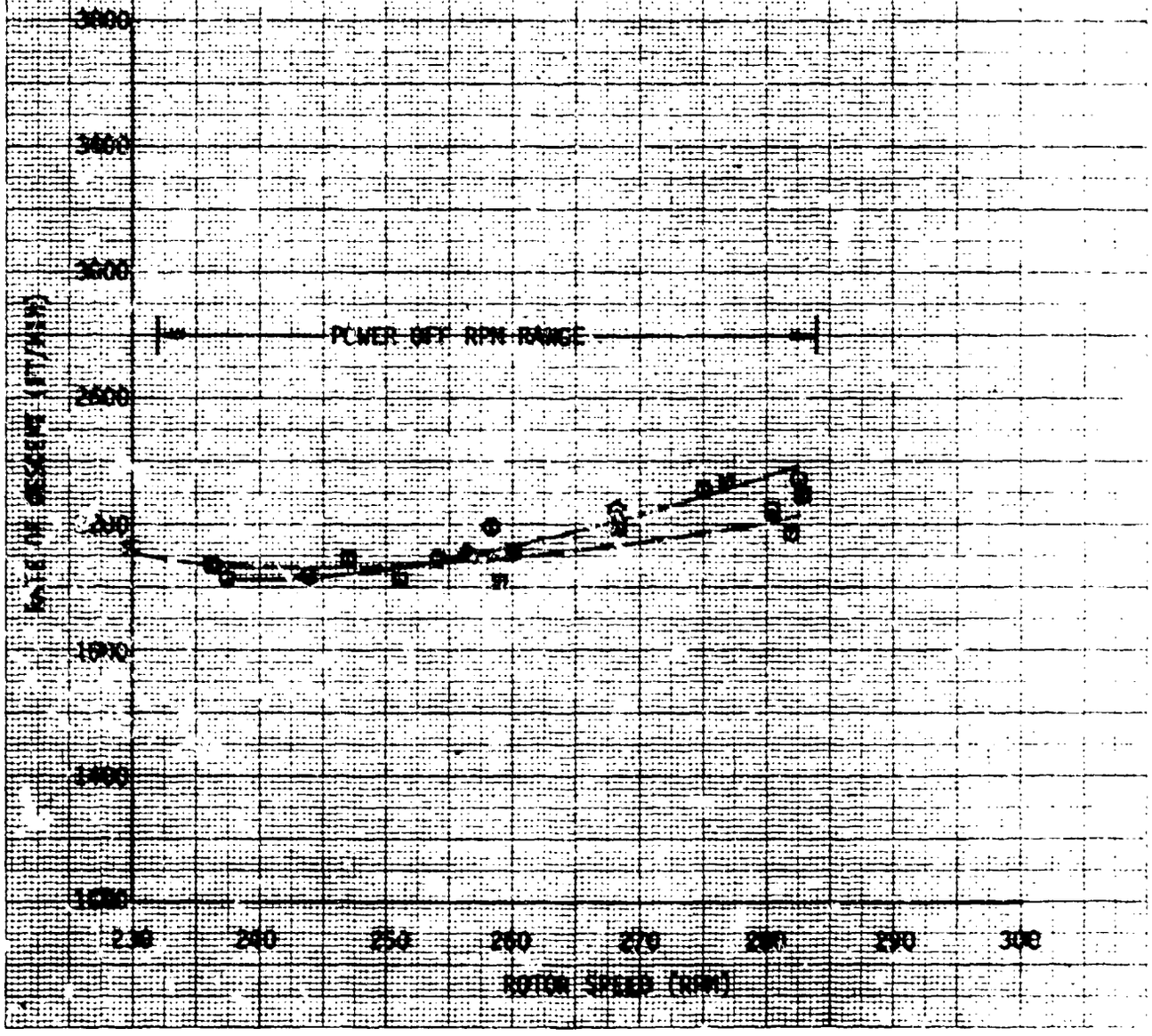


FIGURE 79 --
LIMITS OF CYCLIC CONTROL TRAVEL
[illegible] S/N 77-22718

NOTES:
1. [illegible]-STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
3. NO. 1 AND NO. 2 BOOST SYSTEMS ON
4. MATCHED ARMS DENOTE LIMIT OF CYCLIC CONTROL TRAVEL
5. TOTAL DIRECTIONAL CONTROL TRAVEL EQUALS 4.8 INCHES
6. TOTAL LONGITUDINAL CONTROL TRAVEL EQUALS 9.3 INCHES
7. TOTAL LATERAL CONTROL TRAVEL EQUALS 10.2 INCHES

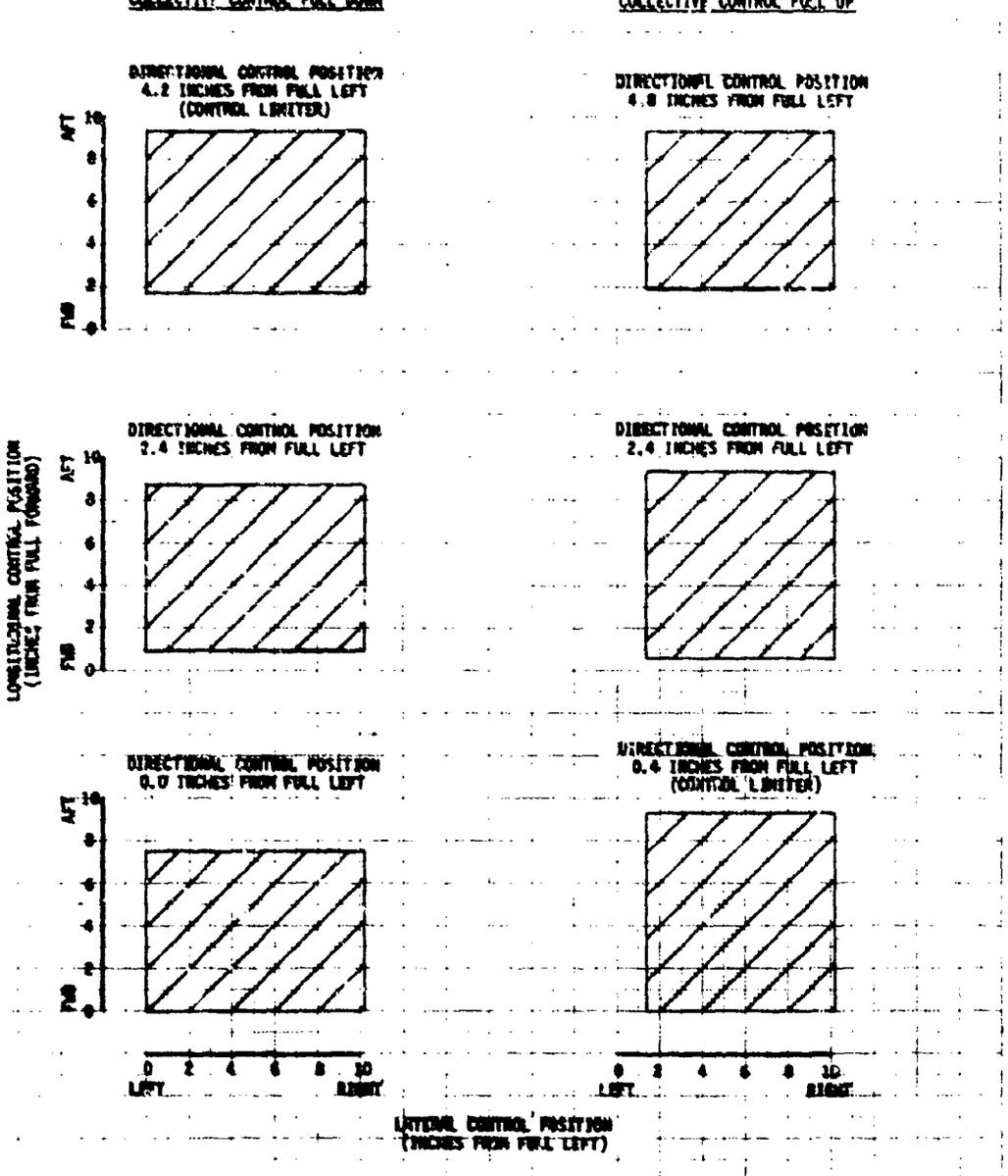


FIGURE 8D

LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS

UH-60A USA S/N 77-22716

NOTES:
1. ROTOR STATIC
2. FORCE AND POSITION MEASURED AT CENTER OF CONTROL GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
4. NO. 1 AND NO. 2 BOOST SYSTEMS ON
5. AFCS/TRIM ON
6. COLLECTIVE CONTROL FULL DOWN
7. DIRECTIONAL CONTROL CENTERED
8. SHADED SYMBOL DENOTES TRIM POSITION
9. TOTAL LONGITUDINAL CONTROL TRAVEL EQUALS 9.3 INCHES

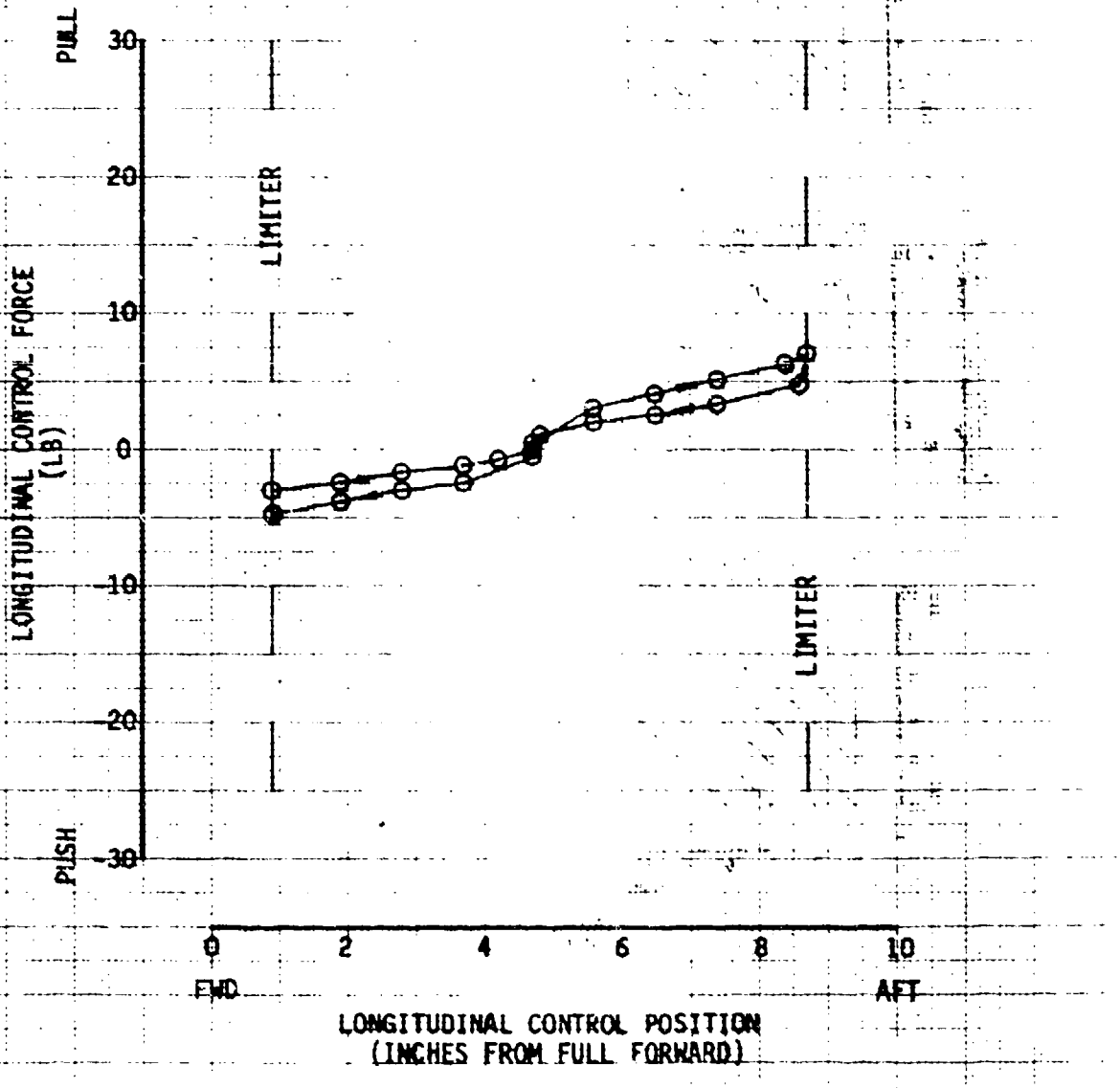


FIGURE 8J
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
UH-60A USA S/N 77-22016

NOTES:
1. ROTOR STATIC
2. FORCE AND POSITION MEASURED AT CENTER OF CONTROL GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
4. NO. 1 AND NO. 2 BOOST SYSTEMS ON
5. AFCS/TRIM ON
6. COLLECTIVE CONTROL FULL UP
7. DIRECTIONAL CONTROL CENTERED
8. SHADED SYMBOL DENOTES TRIM POSITION
9. TOTAL LONGITUDINAL CONTROL TRAVEL EQUALS 9.3 INCHES

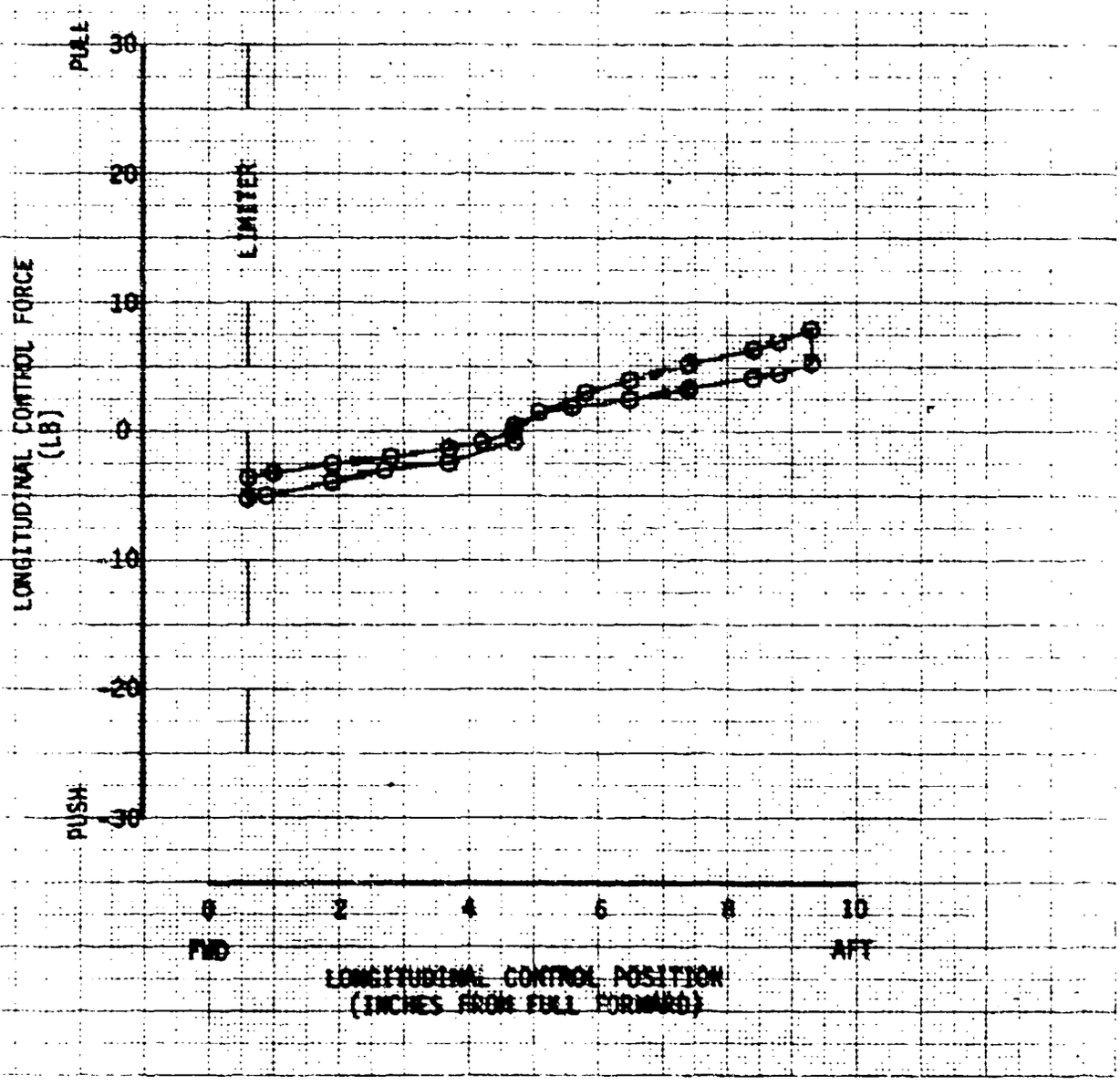


FIGURE 32
LATERAL CONTROL SYSTEM CHARACTERISTICS
UH-60A USA S/N 77-22716

NOTES:
1. ROTOR STATIC
2. FORCE AND POSITION MEASURED AT CENTER OF CONTROL GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
4. NO. 1 AND NO. 2 BOOST SYSTEMS ON
5. AFCS/TRIM ON
6. COLLECTIVE CONTROL FULL DOWN
7. DIRECTIONAL CONTROL CENTERED
8. SHADED SYMBOL DENOTES TRIM POSITION
9. TOTAL LATERAL CONTROL TRAVEL EQUALS 10.2 INCHES

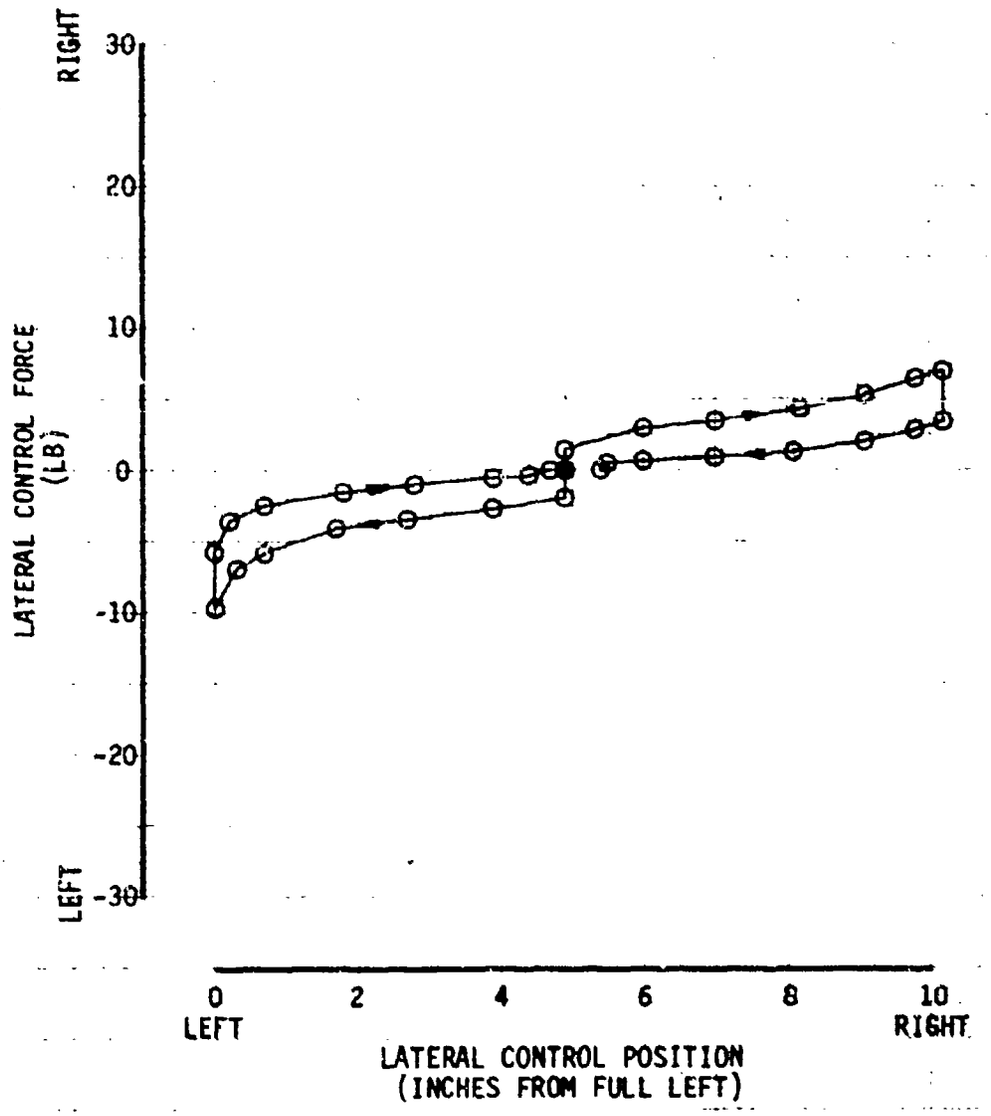


FIGURE 83
LATERAL CONTROL SYSTEM CHARACTERISTICS
UH-60A USA S/N 77-22716

NOTES:
1. ROTOR STATIC
2. FORCE AND POSITION MEASURED AT CENTER OF CONTROL GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
4. NO. 1 AND NO. 2 BOOST SYSTEMS ON
5. AFCS/TRIM ON
6. COLLECTIVE CONTROL FULL UP
7. DIRECTIONAL CONTROL CENTERED
8. SHADED SYMBOL DENOTES TRIM POSITION
9. TOTAL LATERAL CONTROL TRAVEL EQUALS 10.2 INCHES

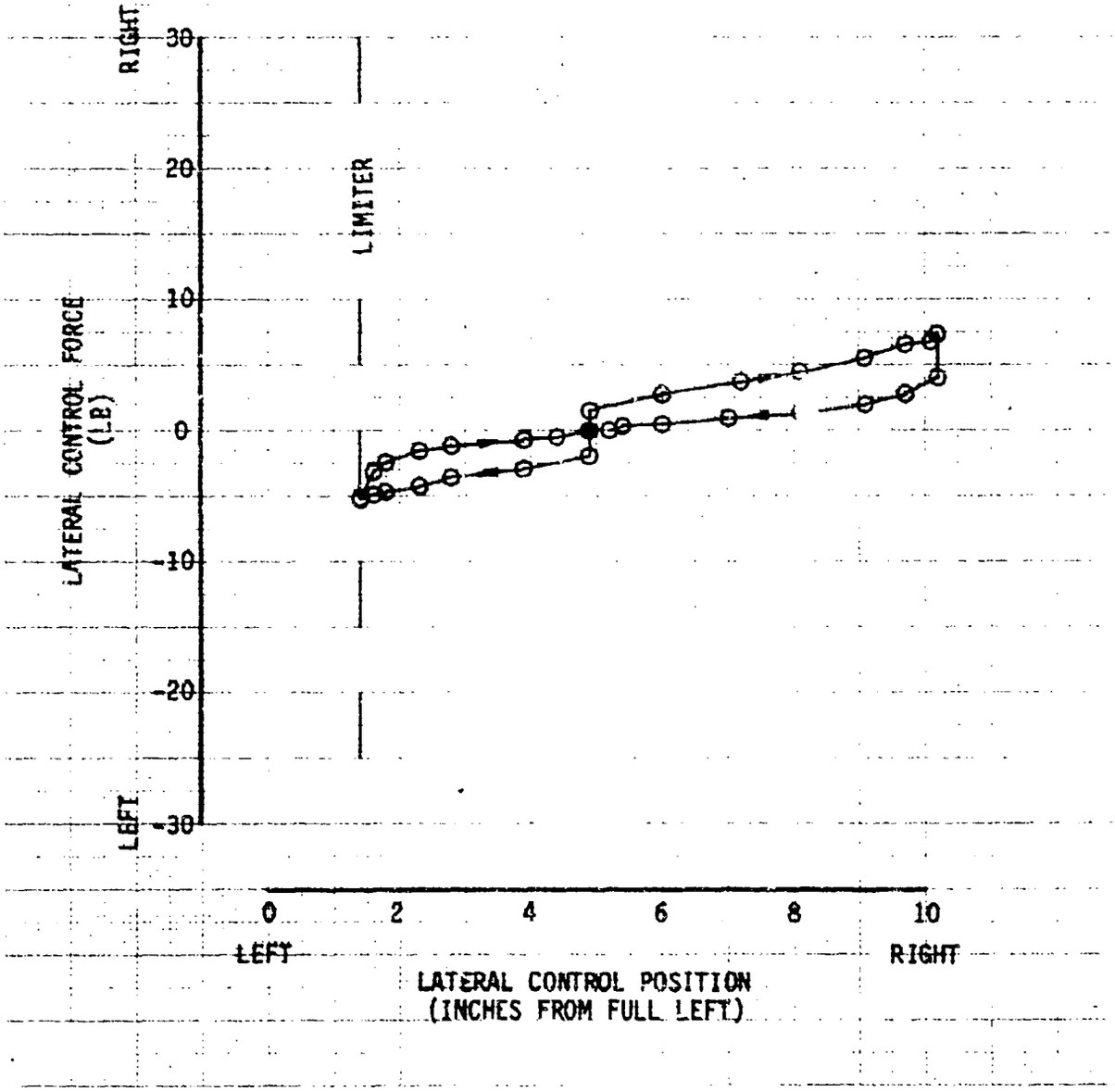
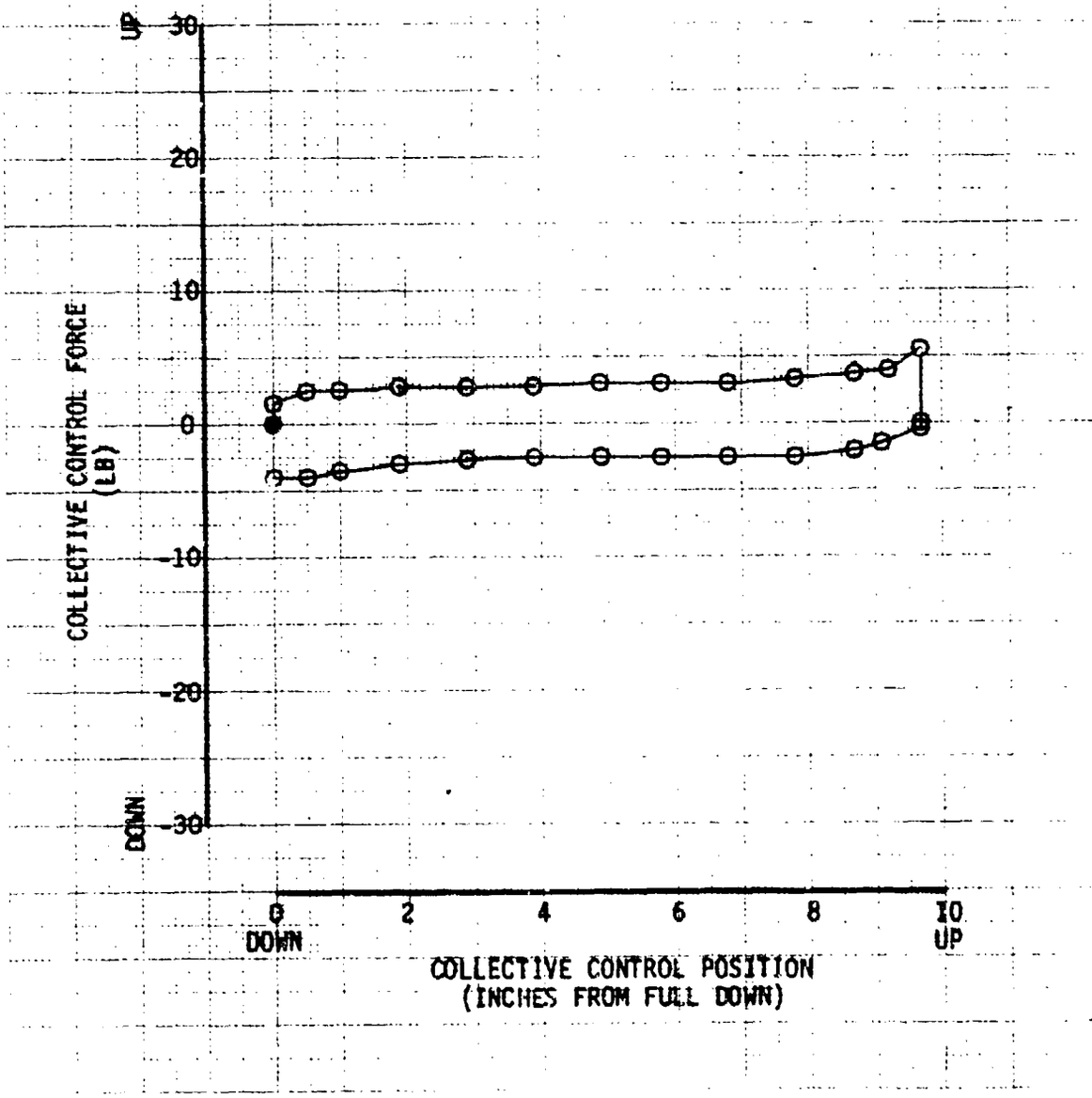


FIGURE 84
COLLECTIVE CONTROL SYSTEM CHARACTERISTICS
UH-60A USA S/N 77-22716

NOTES:
1. ROTOR STATIC
2. FORCE AND POSITION MEASURED AT CENTER OF CONTROL GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
4. NO. 1 AND NO. 2 BOOST SYSTEMS ON
5. AFCS/TRIM ON
6. CYCLIC CONTROL CENTERED
7. DIRECTIONAL CONTROL CENTERED
8. SHADED SYMBOL DENOTES BEGINNING POINT
9. TOTAL COLLECTIVE CONTROL TRAVEL EQUALS 9.7 INCHES
10. ADJUSTABLE FRICTION SET TO ZERO



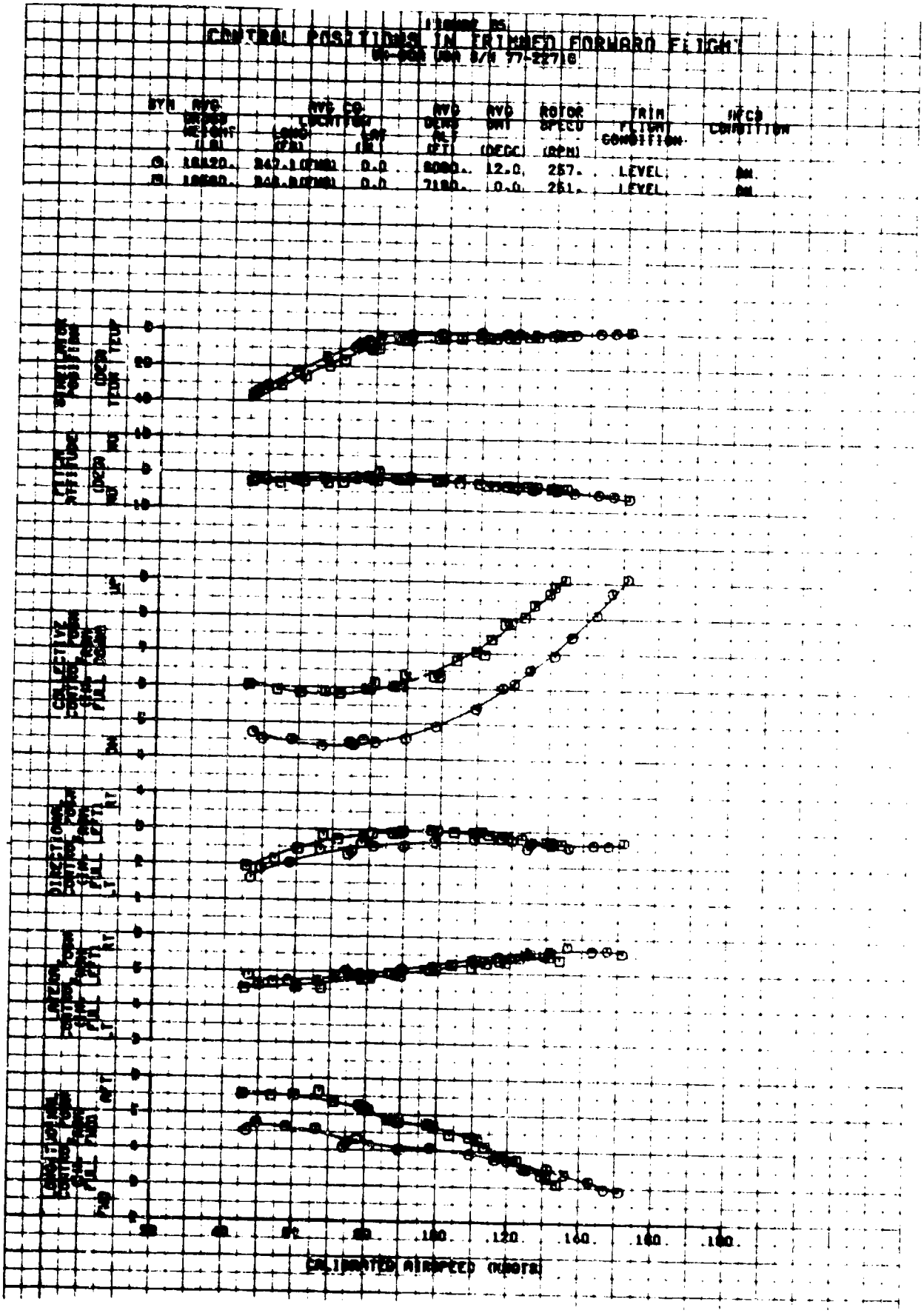

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
SYM AVG GROSS WEIGHT (LB) AVG CG LOCATION LONG (FS) LAT (IN) AVG DENS ALT (FT) AVG OAT (DEG C) ROTOR SPEED (RPM) TRIM FLIGHT CONDITION HFCS CONDITION
O 18420. 247.1(RPM) 0.0 8000. 12.0 257. LEVEL ON
□ 18680. 248.8(RPM) 0.0 7180. 0.0 251. LEVEL ON
COLLECTIVE
LATERAL
DIRECTIONAL
CALIBRATED AIRSPEED (KNOTS)

FIGURE 36.

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

UH-60A (USA S/N 77-22716)

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONFIGURATION
○	18420.	347.1 (FWD)	0.0	6000.	12.0	257.	LEVEL	ON
□	17180.	348.8 (FWD)	0.0	4500.	11.5	257.	LEVEL	ON

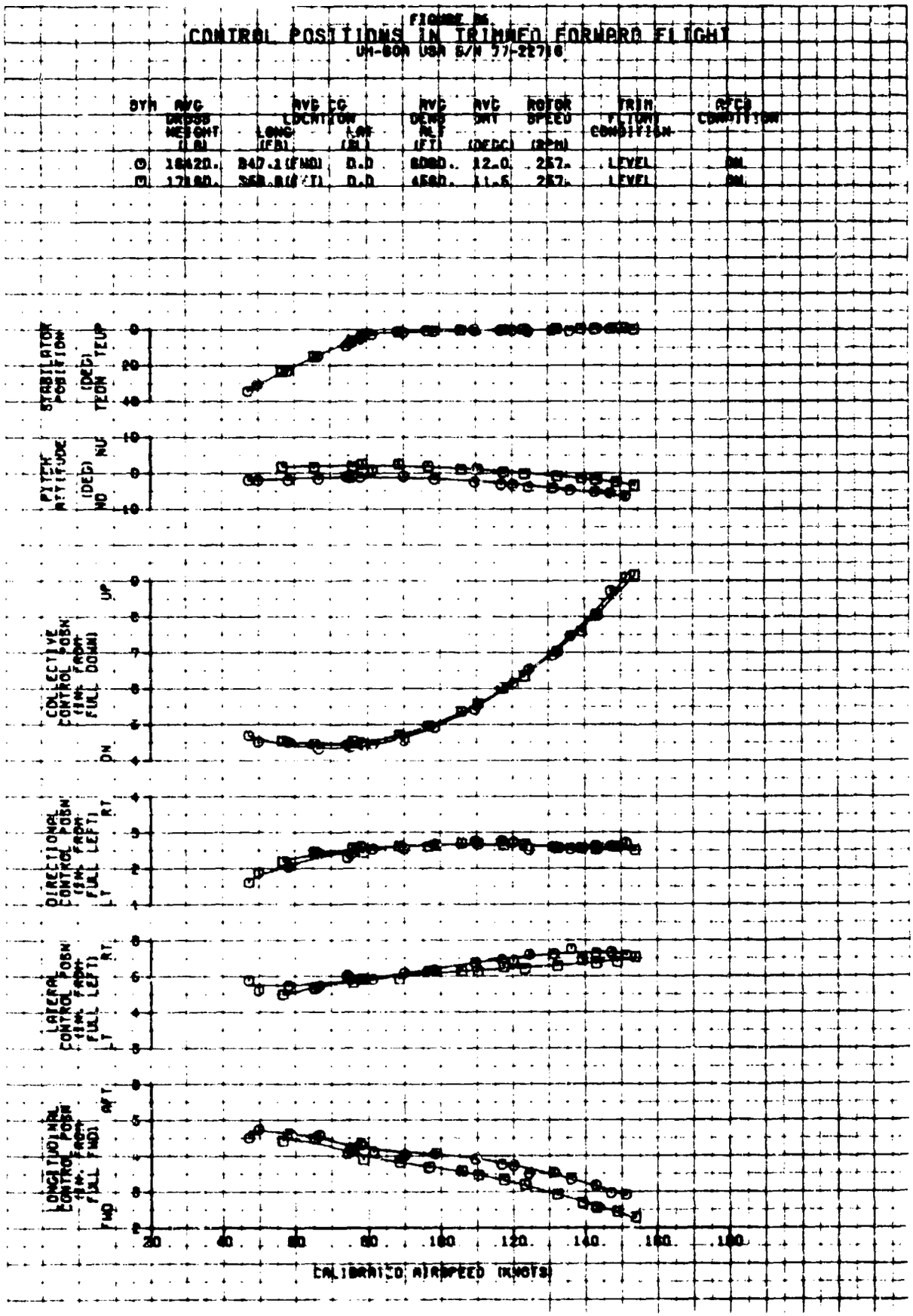


FIGURE 67

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

UH-60A (BSA S/N 77-22710

SYM	AVD GROSS WEIGHT (LB)	AVD CG LOCATION LONG (FS)	LAT (IN)	AVD DENSITY ALT (FT)	AVD OAT (DEGC)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	REMARKS CONFIGURATION
⊙	16000.	348.8	AFT 0.0	8000.	12.6	257.	LEVEL	BSL

NOTE: CARGO DOORS AND GUNNER WINDOWS FULLY OPEN

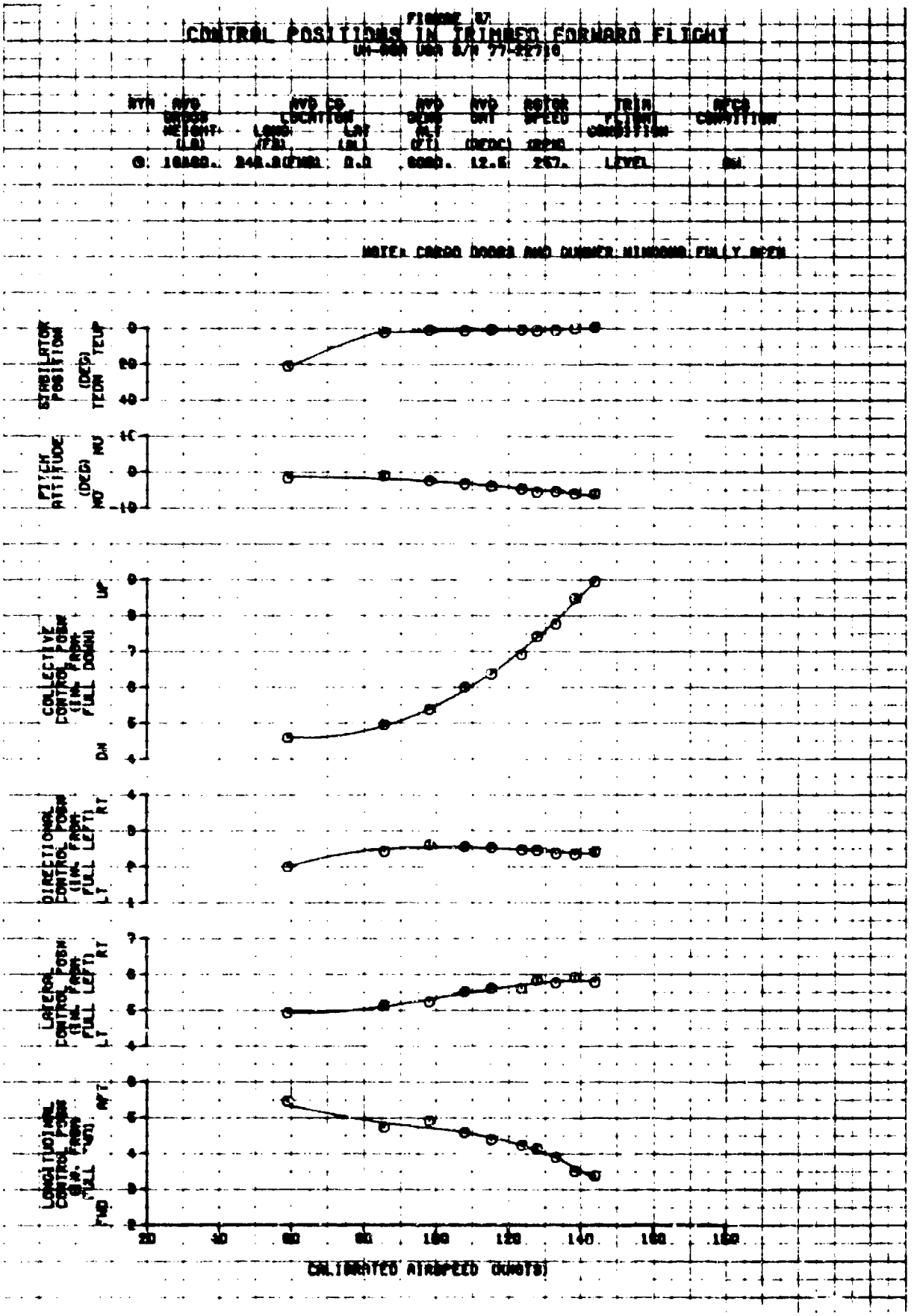


FIGURE 88.

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LBS)	AVG CG LOCATION LONG. (FS)	AVG CG LOCATION LAT. (IN.)	AVG DENS ALT (FT)	AVG OAT (DEGC)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONDITION
○	18820.	347.2(FWD)	0.0	8420.	11.0	258.	LEVEL	ON
□	18780.	342.1(FWD)	0.0	8740.	18.6	259.	LEVEL	ON

NOTE: CONFIGURED WITH 2 INFRARED SUPPRESSORS, XM-130, AND AN/ALQ-144.

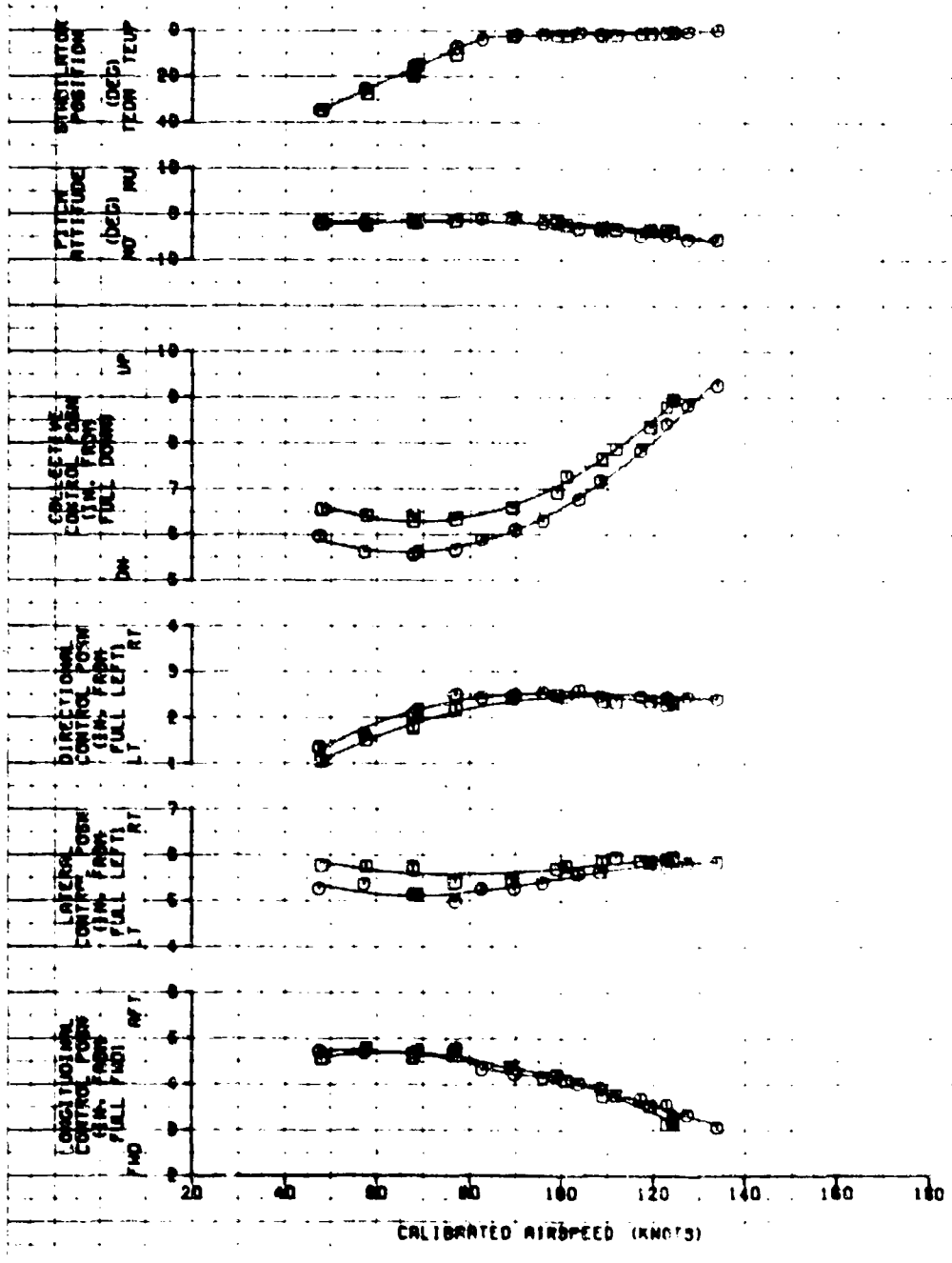
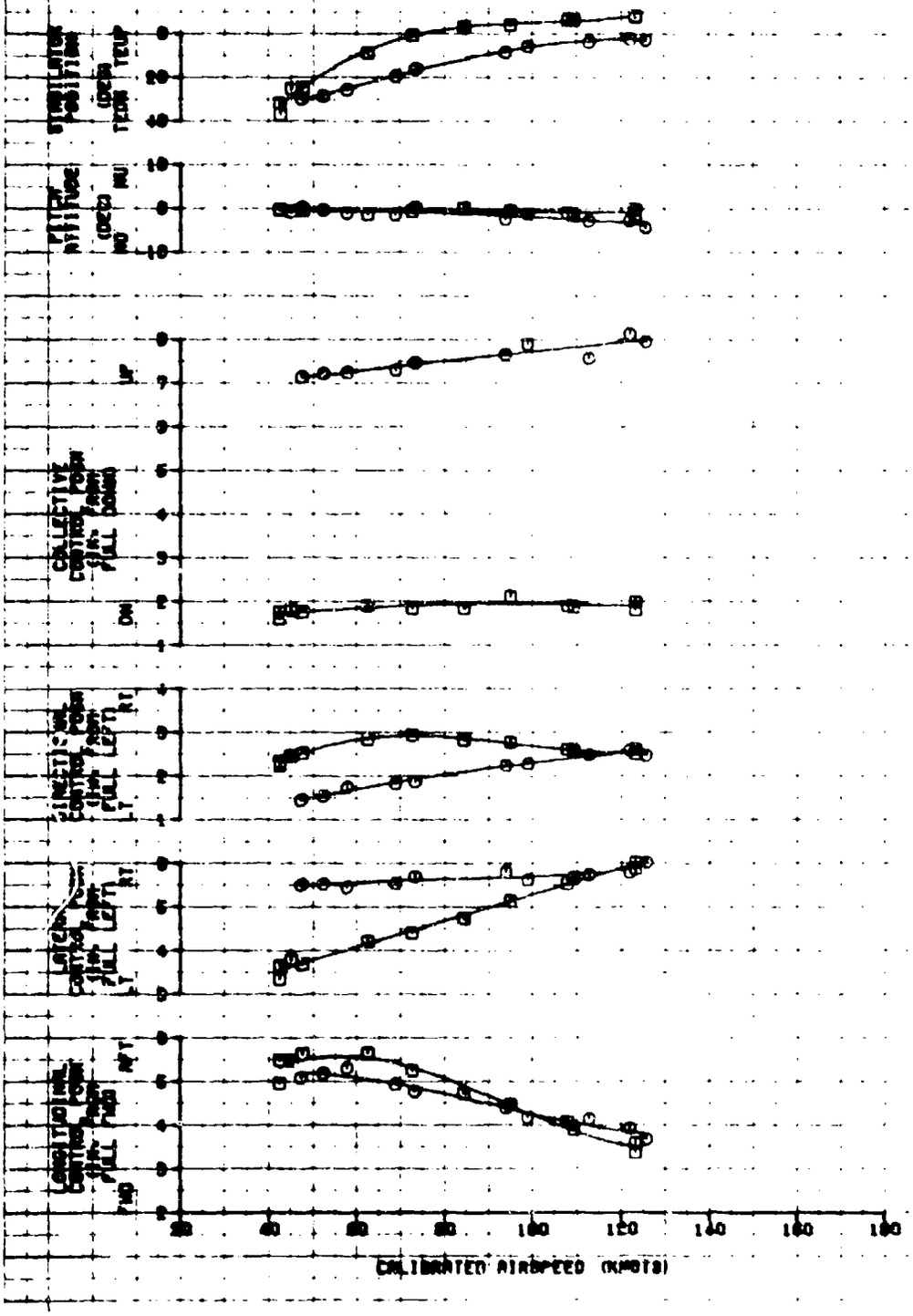


FIGURE 89

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

UH-60A (GEN 8/N 77-22716

SYM	GROSS WEIGHT (LBS)	CG LOCATION LONG (FS)	CG LOCATION LAT (IN)	DENSITY ALT (FT)	OAT (DEGC)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONDITION
○	16000.	342.1(STA)	0.0	7200.	12.5	258.	MCP CLIMB	ON
□	16000.	342.2(STA)	0.0	7200.	10.0	257.	AUTOROTATION	ON



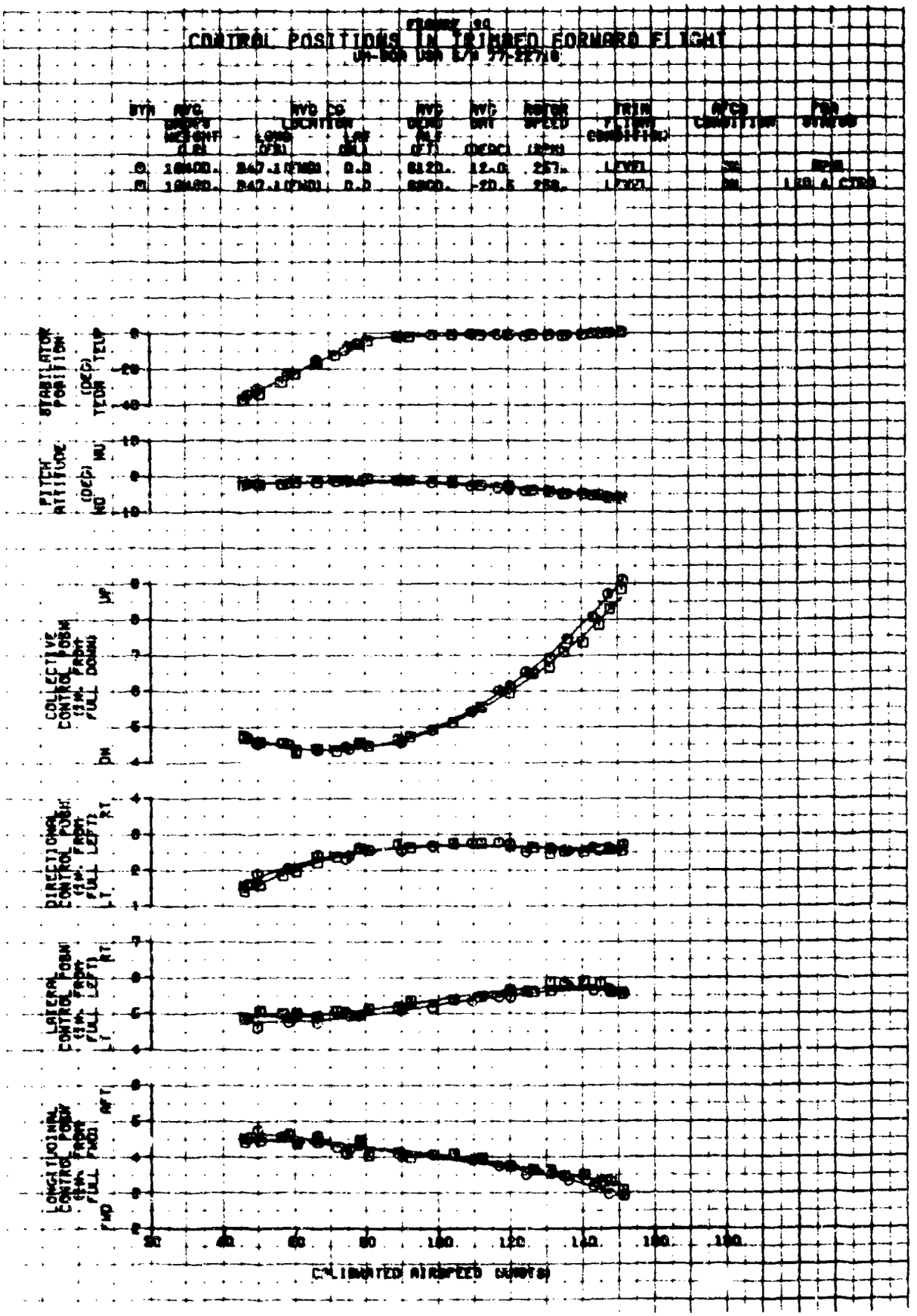


FIGURE 99
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
UH-60A USN S/N 77-22718
STABILATOR POSITION (DEG) TRIM TEMP
PITCH ATTITUDE (DEG) NU
COLLECTIVE CONTROL POSN (IN. FROM FULL DOWN) UP DN
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT) RT LT
LATERAL CONTROL POSN (IN. FROM FULL LEFT) RT LT
LONGITUDINAL CONTROL POSN (IN. FROM FULL AFT) AFT FWD
CALIBRATED AIRSPEED (KNOTS)

FIGURE 31
COLLECTIVE FIXED STATIC LONGITUDINAL STABILITY
UH-60A USA S/N 77-22716

STA	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IML)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
●	15,180	360.8(AFT)	0.1(RT)	7800	28.0	258	LEVEL
■	15,260	360.3(AFT)	0.1(RT)	7980	27.6	258	LEVEL

NOTES: 1. PBA LOCKED IN CENTER POSITION
2. DASHED LINES DENOTE FAIRINGS FROM COMPARABLE DATA OBTAINED FROM GOVERNMENT COMPETITIVE TEST REPORT NO. 74-06-1, FIGURES 64 AND 65; PBA OPERATIONAL
3. NORMAL UTILITY CONFIGURATION

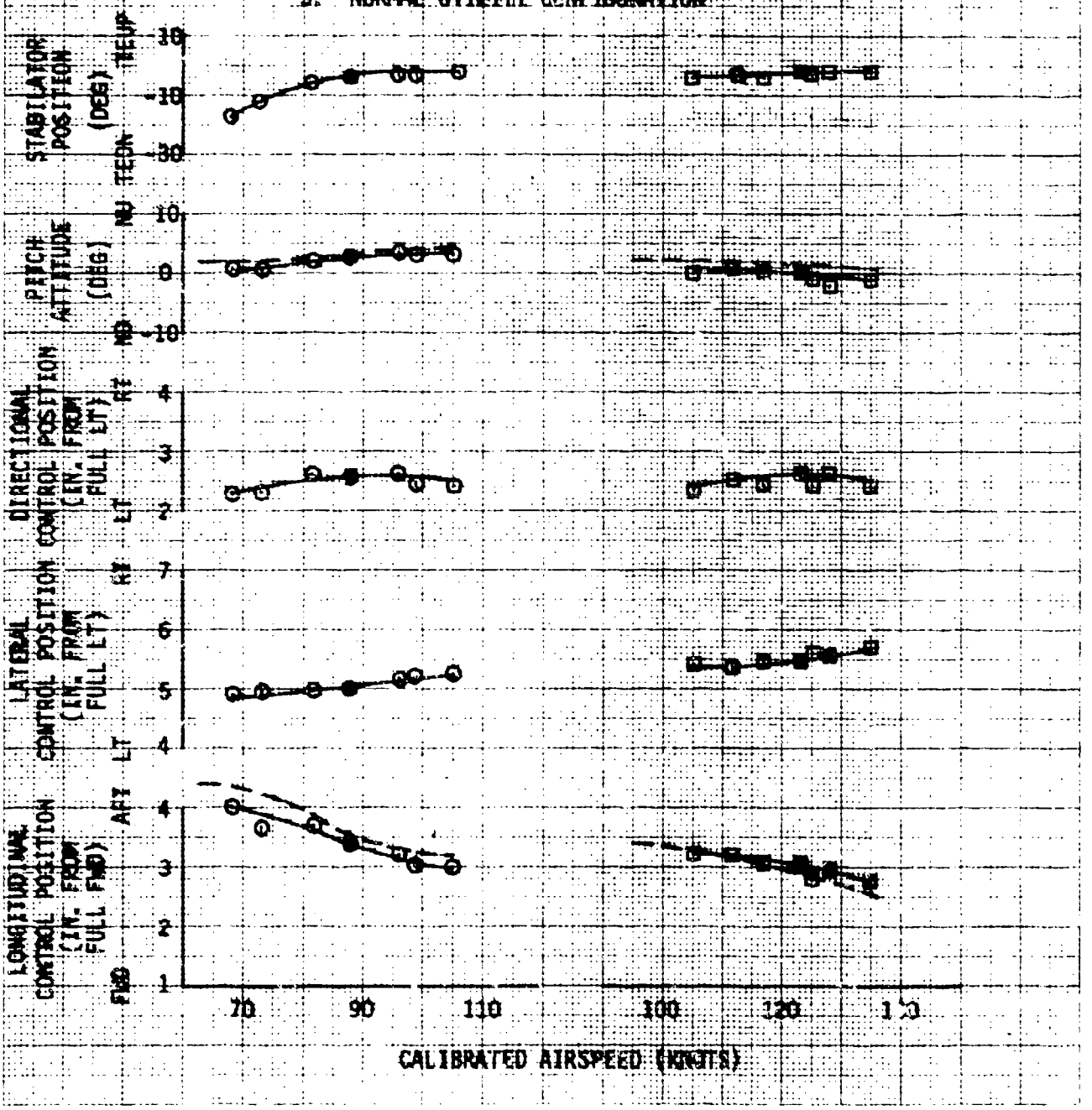


FIGURE 92
COLLECTIVE FIXED STATIC LONGITUDINAL STABILITY
UH-60A USA S/N 77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
16,640	359.5(AFT)	0.1(RT)	7620	27.5	258	LEVEL

NOTE: PWR LOCKED IN CENTER POSITION, NORMAL UTILITY CONFIGURATION

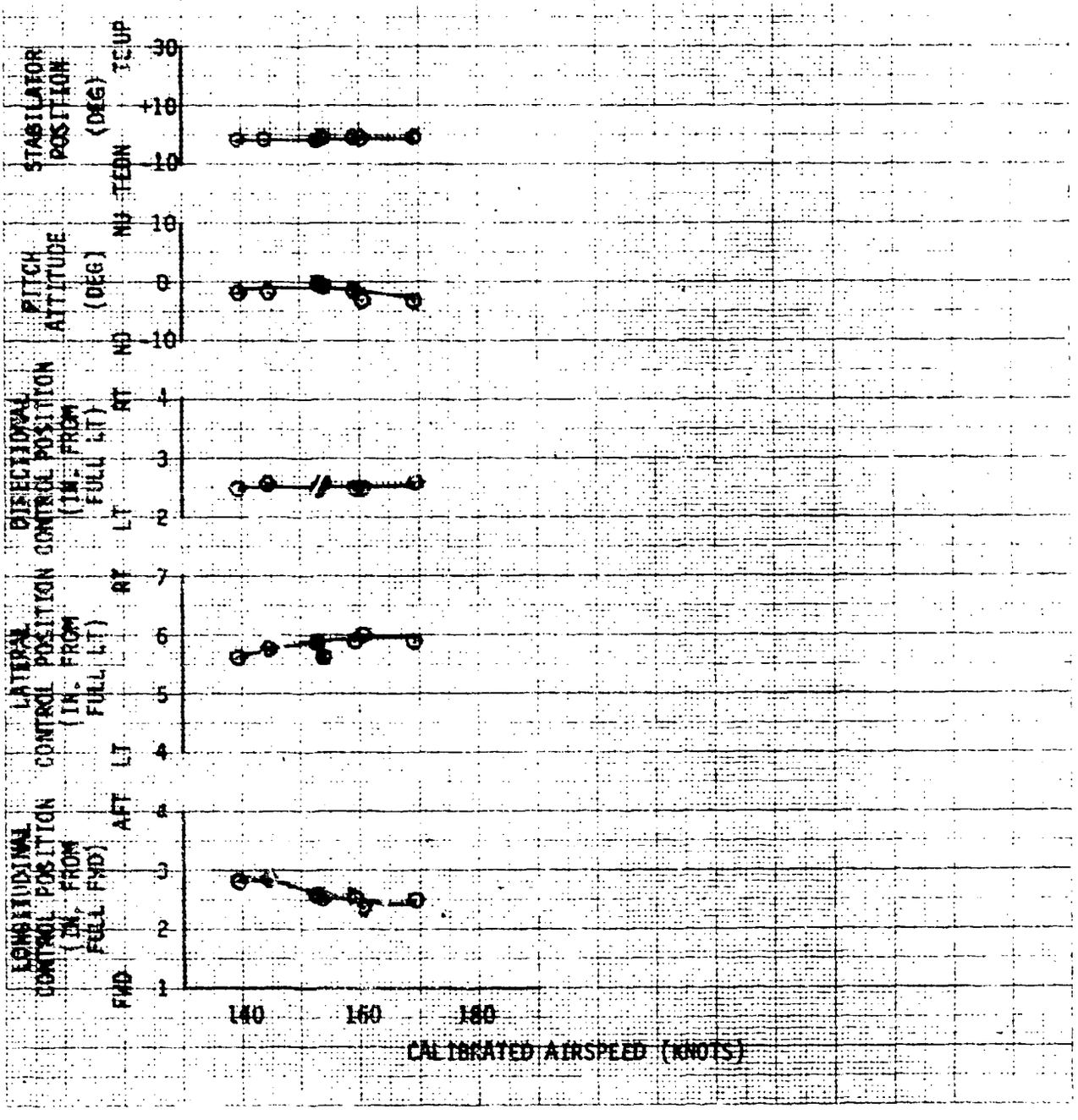


FIGURE 93
MANEUVERING STABILITY
OH-6OA USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LBS)	AVG LOCATION LONG (FS)	AVG LOCATION LAT (BL)	AVG HD (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
○	16,240	359.0(AFT)	0.2(RT)	6500	11.0	258	118	RT TURN
□	16,720	359.0(AFT)	0.2(RT)	6520	11.0	258	118	LT TURN

NOTES: 1. SHADED SYMBOLS DENOTE TRIM POINT
2. NORMAL UTILITY CONFIGURATION

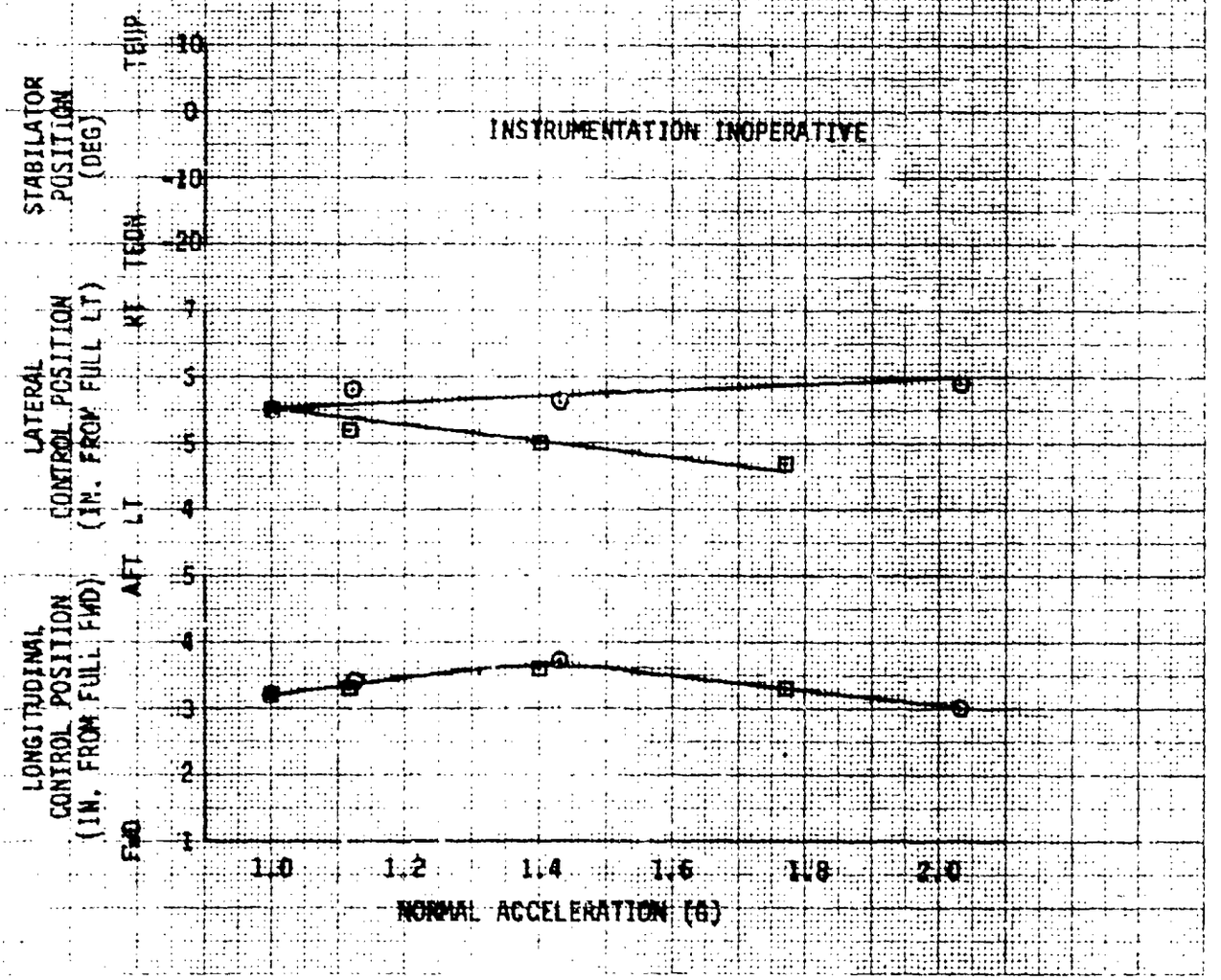


FIGURE 94
MANEUVERING STABILITY
UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
◇	15,420	359.5(AFT)	0.2(RT)	8500	25.0	258	130	RT TURN
□	15,380	359.5(AFT)	0.2(RT)	8320	25.0	258	132	LT TURN

NOTES: 1. PBA LOCKED IN CENTER POSITION
2. SHADED SYMBOLS DENOTE TRIM POINT
3. NORMAL UTILITY CONFIGURATION

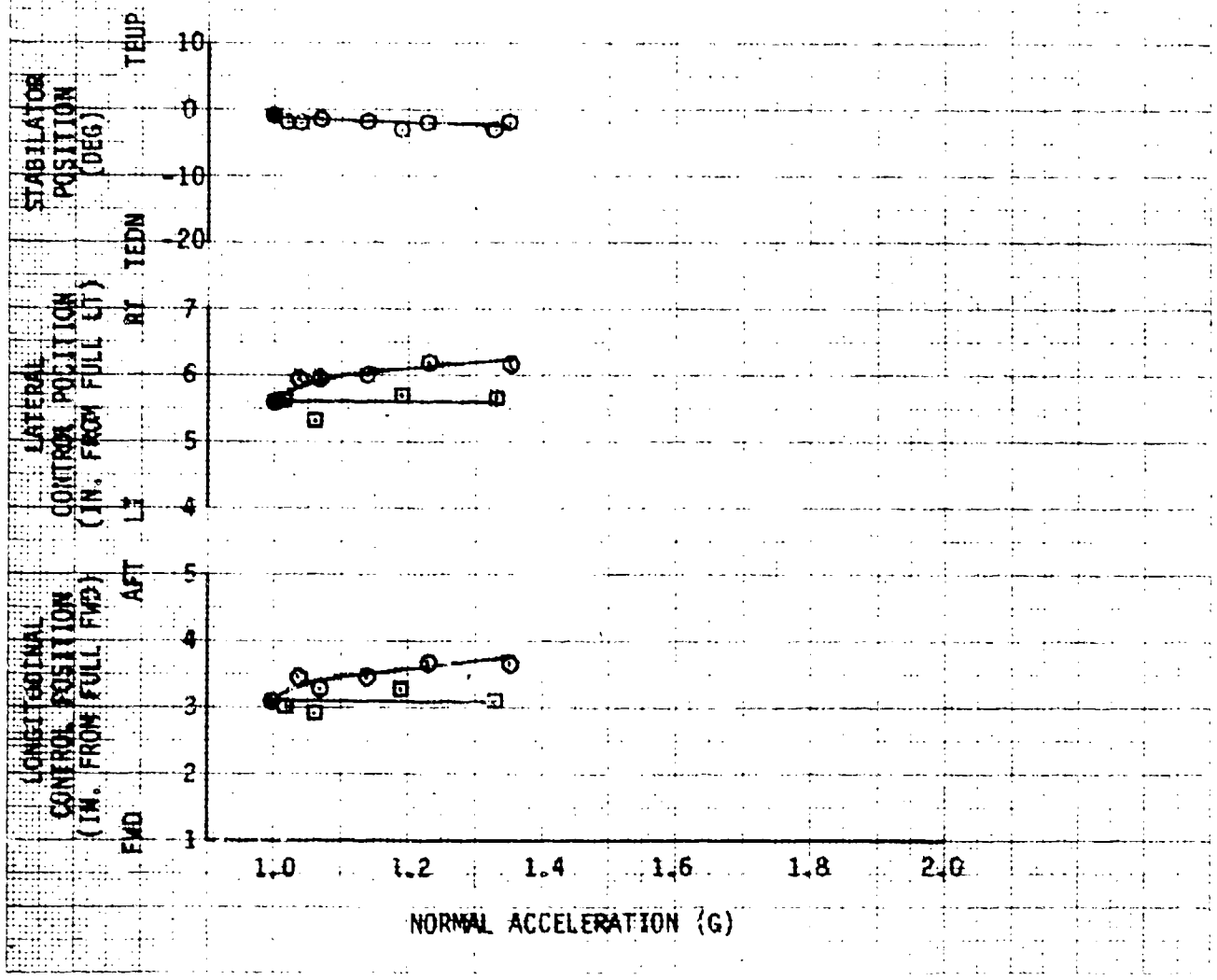


FIGURE 95
LONGITUDINAL CONTROLLABILITY
UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG H_D (FT)	AVG OAT (° C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)
⊙	19,920	360.2(AFT)	0.2(RT)	6420	11.0	258	90
⊡	19,180	360.1(AFT)	0.2(RT)	6780	11.0	257	140

NOTES: 1. CONFIGURATION - NORMAL UTILITY
2. TRIM LEVEL FLIGHT

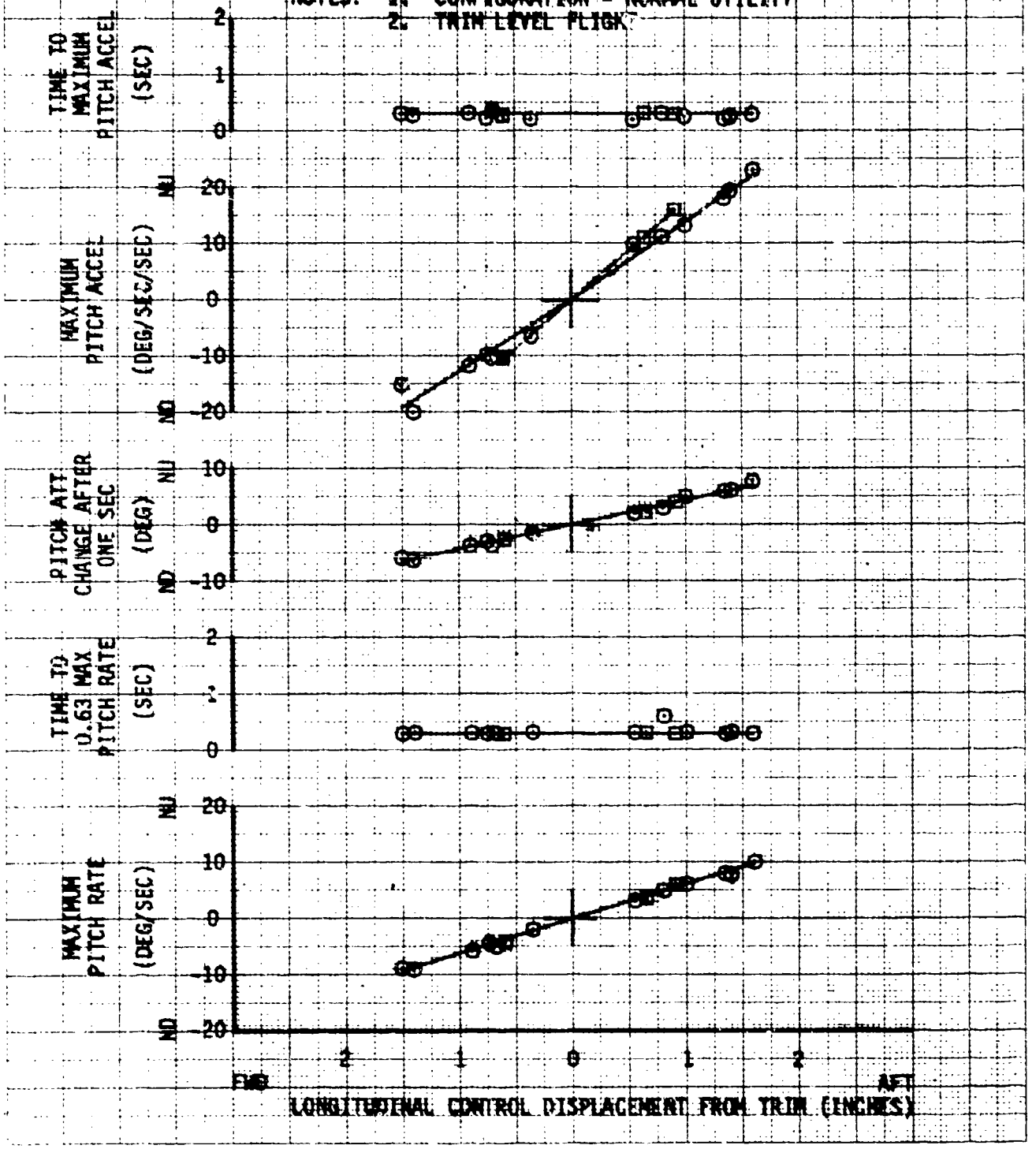


FIGURE
PUSHOVER MANEUVER
UH-60A USA S/N 77-22716

SOLID LINE	SHORT DASH	LONG DASH	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KTS)
			20280	347.5 (FWD)	0.0	3840	18.5	258	106

CONFIGURATION: NORMAL UTILITY

CONTROL POSITION (IN. FROM FULL DOWN) UP / DN
BOOM SYSTEM CALIBRATED AIRSPEED (KNOTS)
NORMAL ACCELERATION (G'S)

RATES
PITCH (DEG/SEC) NU / ND
ROLL (DEG/SEC) RT / LT
YAW (DEG/SEC) RT / LT

ATTITUDES
PITCH (DEGREES) NU / ND
ROLL (DEGREES) RT / LT
YAW (DEGREES) RT / LT

LONGITUDINAL (IN. FROM FULL FWD) AFT / FWD
LATERAL (IN. FROM FULL LEFT) RT / LT
DIRECTIONAL (IN. FROM FULL LEFT) RT / LT

TIME - (SECONDS)

FIGURE 97
LATERAL CONTROLLABILITY
UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (BPA)	LAT. (BL)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
⊙	19,320	359.1(AFT)	0.2(RT)	6500	11.0	258	90	LEVEL
⊡	19,000	359.1(AFT)	0.2(RT)	6700	11.0	258	140	LEVEL

NOTE: CONFIGURATION - NORMAL UTILITY

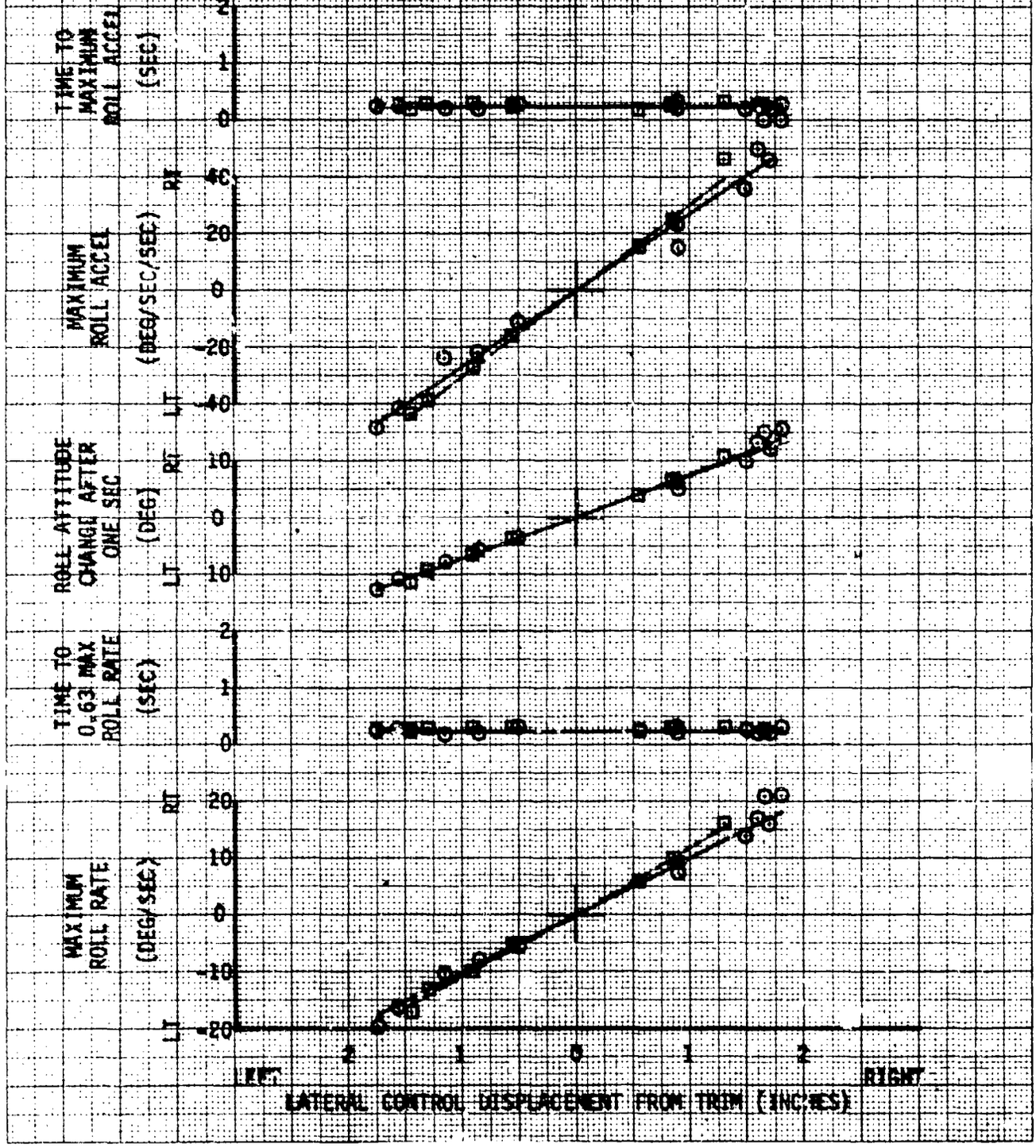


FIGURE 98
RIGHT LATERAL STEP
UH-60A USA S/N 76-22718

SOLID LINE	SHORT DASH	LONG DASH	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KTS)	CONFIGURATION
			19580	359.1 (AFT)	-0.2 (LT)	6400	11.5	258	90	NORMAL UTIL

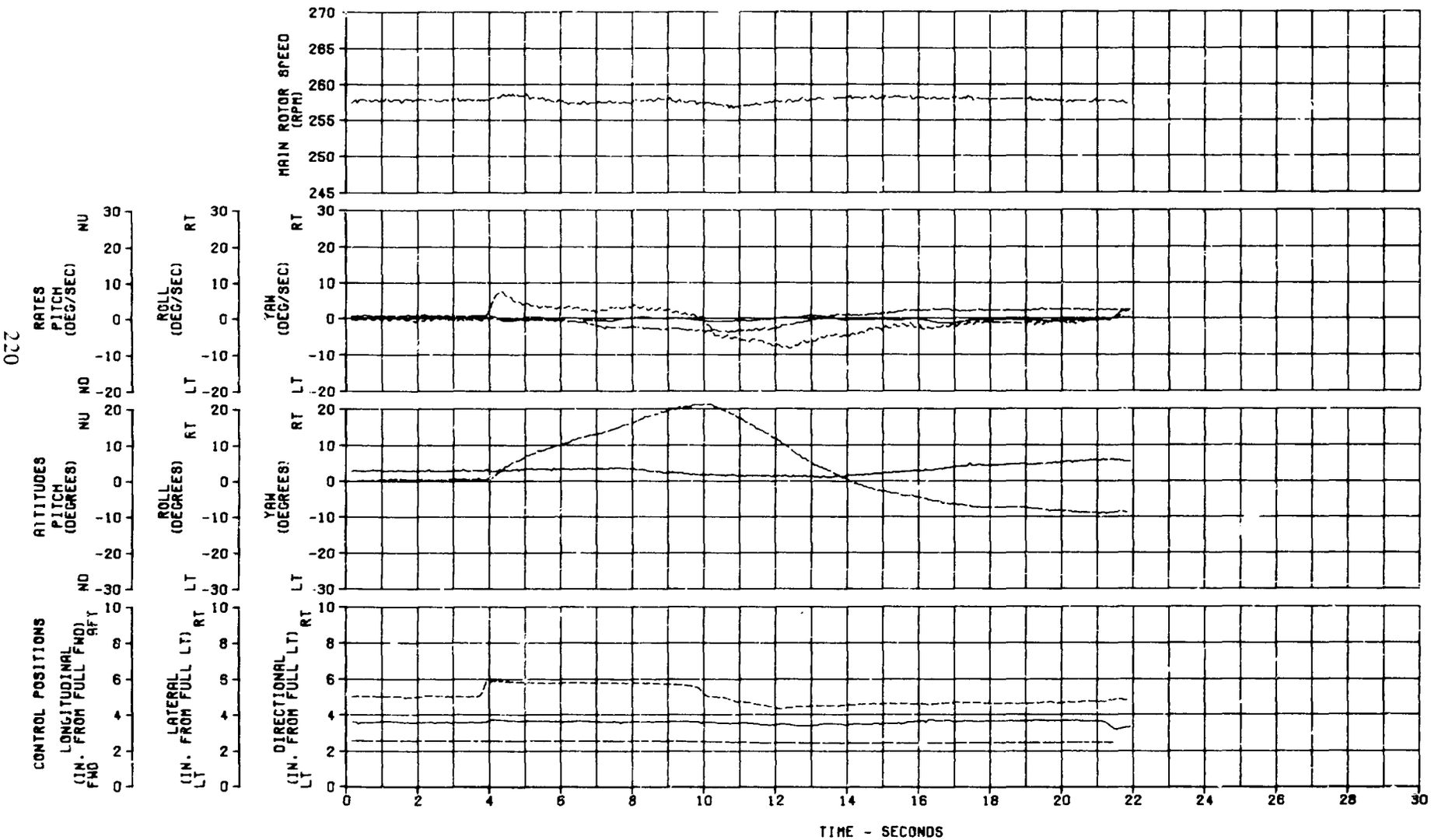


FIGURE 99
DIRECTIONAL CONTROLLABILITY
UH-60A USA S/N 77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (R)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
⊙	19,380	359.1(AFT)	0.2(RT)	6500	11.0	2	90	LEVEL
⊡	18,680	358.7(AFT)	0.2(RT)	6740	11.0	258	140	LEVEL

NOTES: 1. CONFIGURATION - NORMAL UTILITY

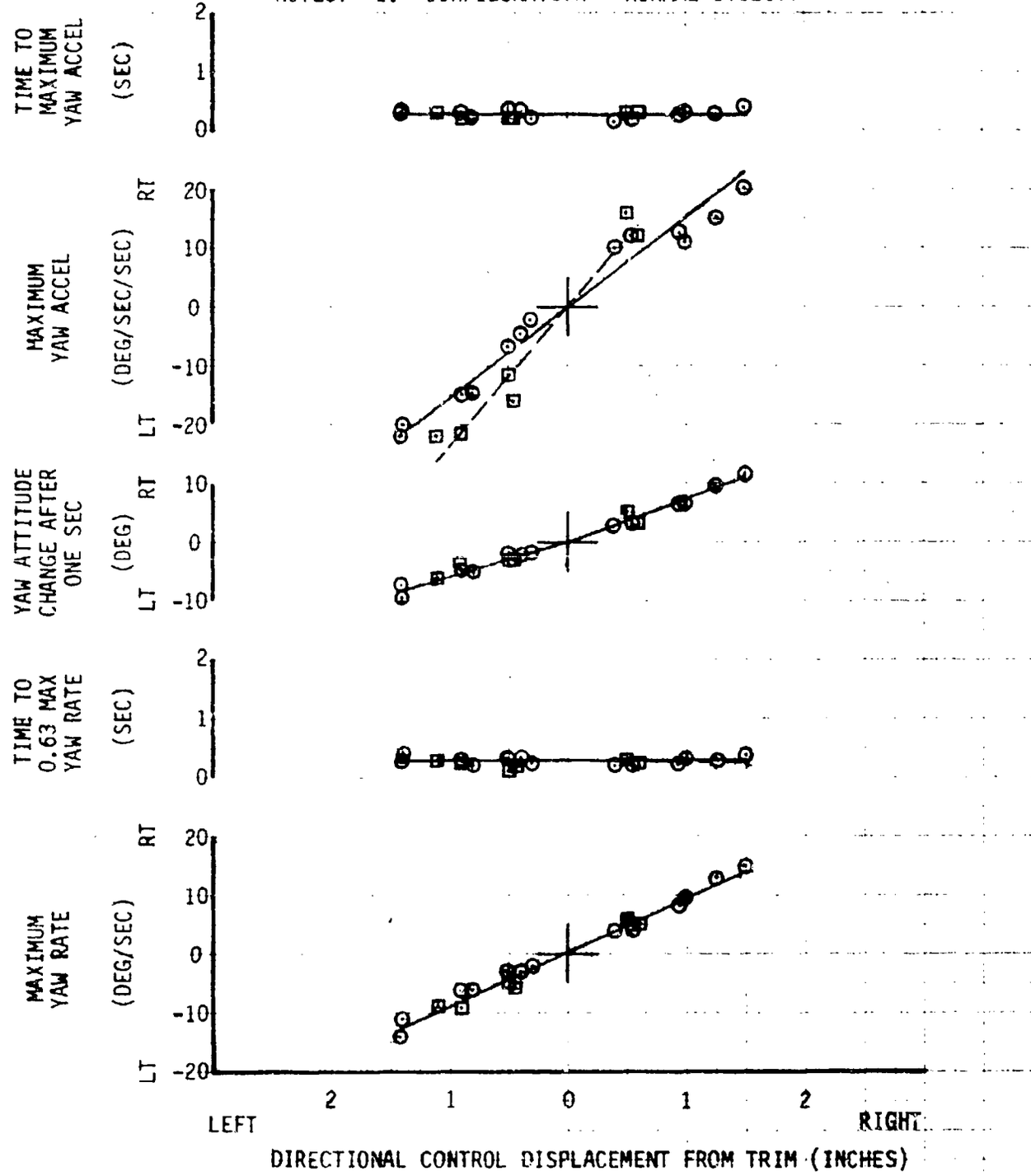


FIGURE 109

TAKEOFF CHARACTERISTICS

UH-60A USA S/N 77-22716

SHORT	LONG	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	TRIM ROTOR SPEED (RPM)	STABILATOR MODE	TAKEOFF PROFILE
OASM	OASM	15620	345.6 (FWD)	-0.1 (LT)	3540	23.0	258	AUTOMATIC LONGITUDINAL CONTROL FIXED	

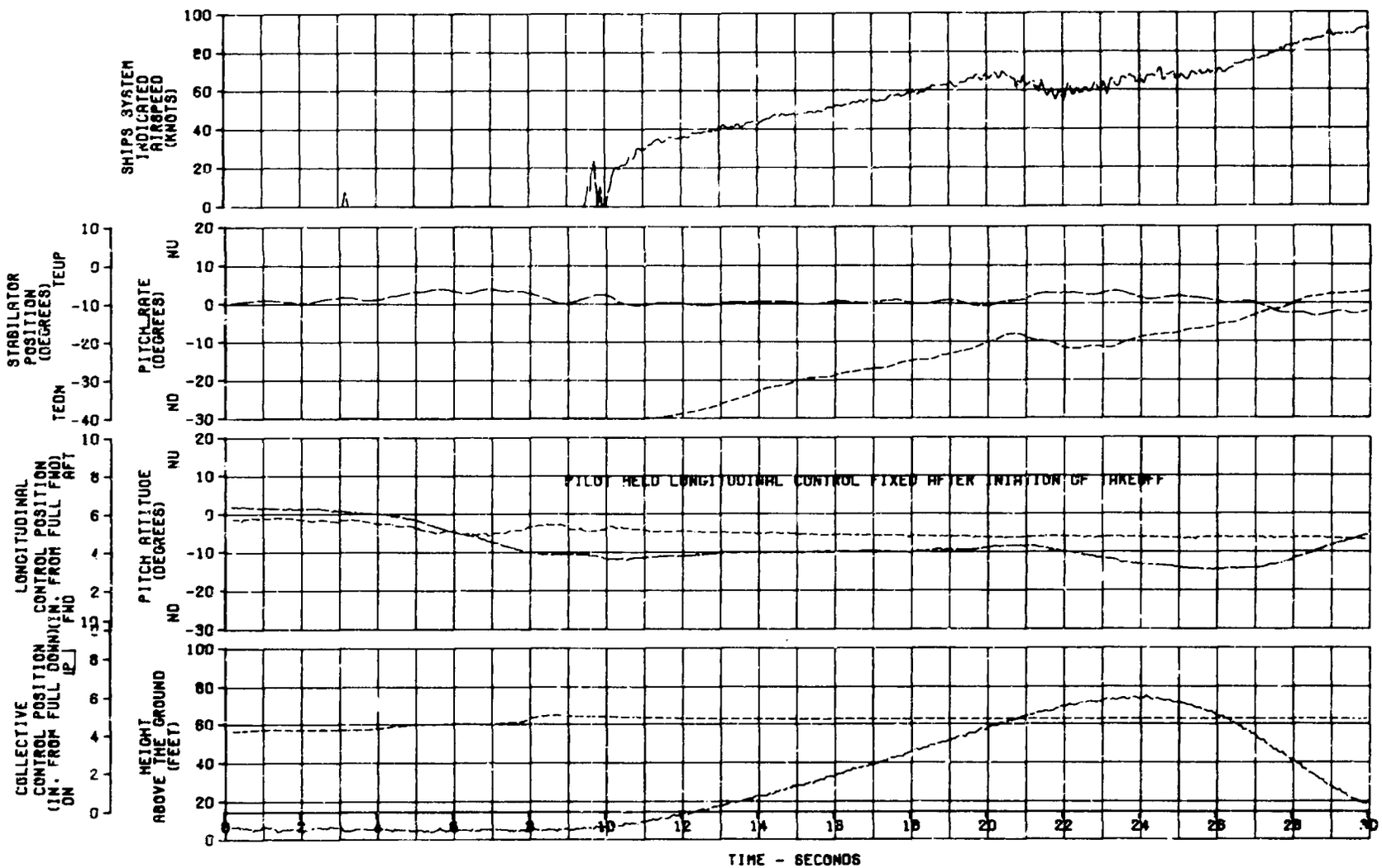
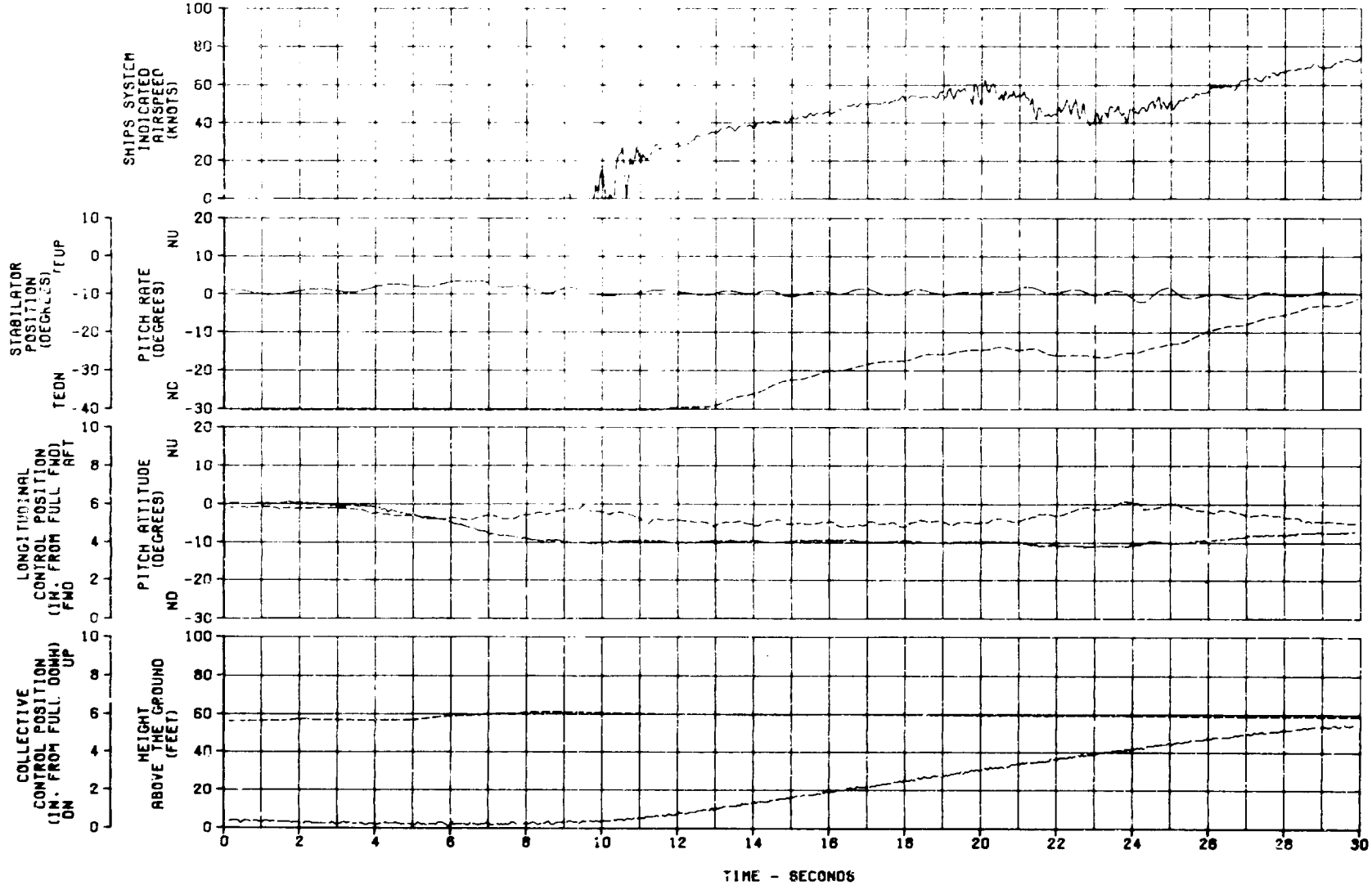


FIGURE 3-101

TAKEOFF CHARACTERISTICS

UH-60A USA S/N 77-22716

SHORT DASH	LONG DASH	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	TRIM ROTOR SPEED (RPM)	STABILATOR MODE	TAKEOFF PROFILE
		15760	345.6 (FWD)	-0.1	3520	22.0		AUTOMATIC	NORMAL



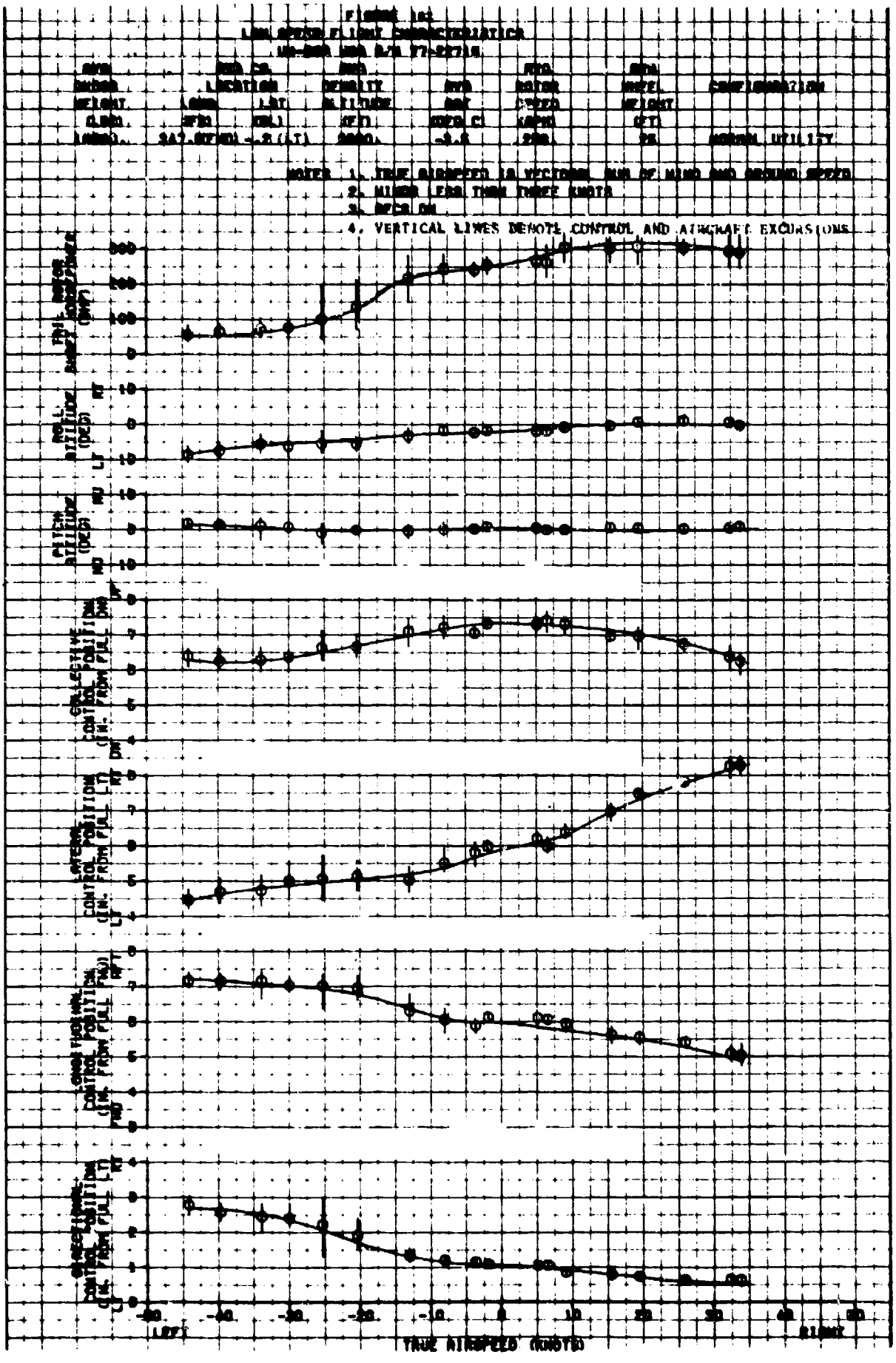

NOTES 1. TRUE AIRSPEED IS VECTORIAL SUM OF WIND AND GROUND SPEED
2. WINDS LESS THAN THREE KNOTS
3. RPCS ON
4. VERTICAL LINES DENOTE CONTROL AND AIRCRAFT EXCURSIONS
TRUE AIRSPEED (KNOTS)
LEFT
RIGHT

FIGURE 30

LOW SPEED FORWARD AND REARWARD FLIGHT CHARACTERISTICS

UH-60A USA S/N 77-22718

AVG. GROSS WEIGHT (LBS)	AVG CG LOCATION LONGL (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG WHEEL HEIGHT (FT)	CONFIGURATION
17140.	347.8(FWD)	-.2 (LT)	2720.	-5.0	258.	25.	NORMAL UTILITY

NOTES: 1. TRUE AIRSPEED IS VECTORIAL SUM OF WIND AND GROUND SPEED.
2. WINDS LESS THAN THREE KNOTS
3. AFCS ON
4. [illegible] AND AIRCRAFT EXCURSIONS

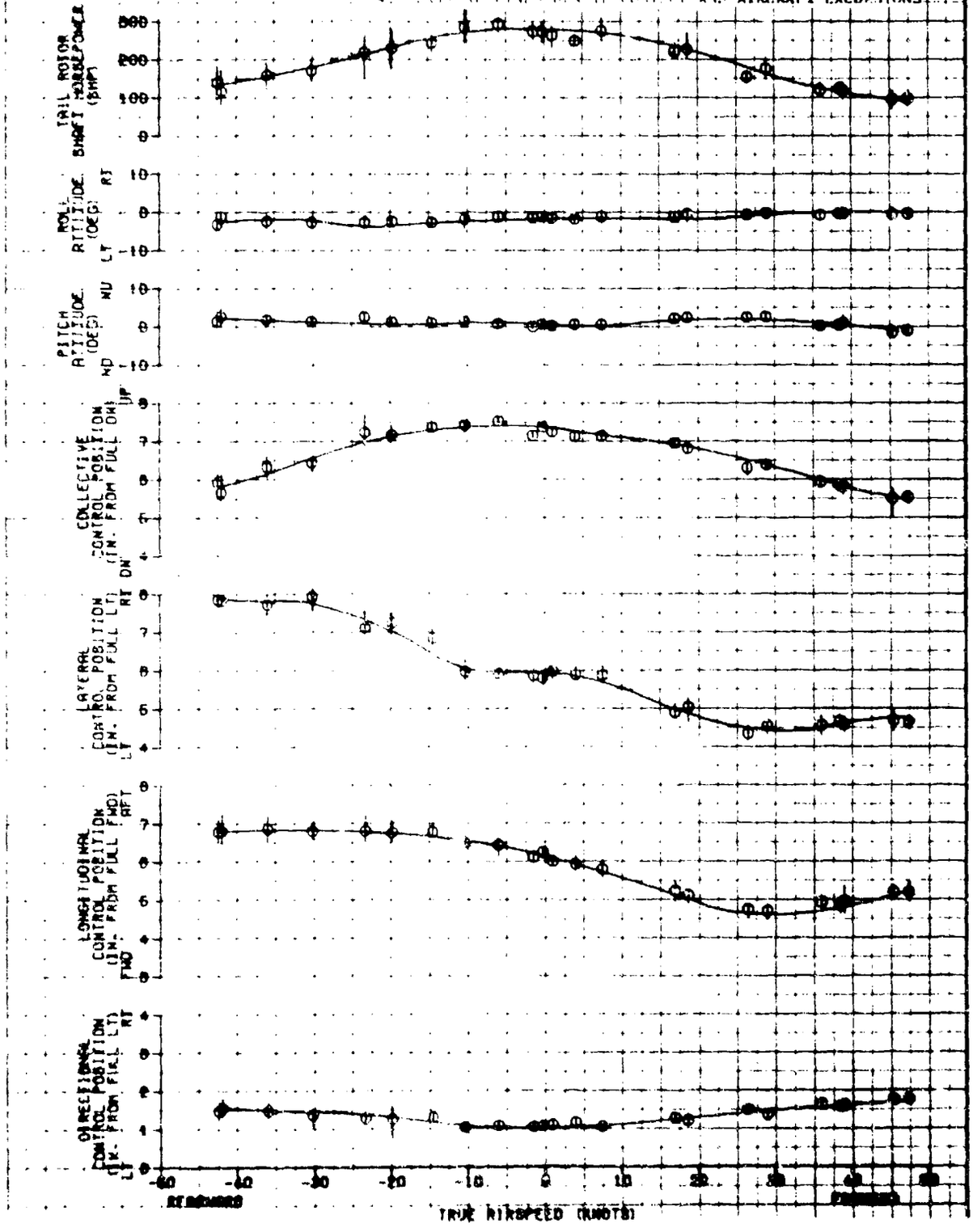


FIGURE 104

LONGITUDINAL FLIGHT CHARACTERISTICS

MD-500E MOD DATA 77-02717

GROSS WEIGHT (LBS)	C.G. LOCATION LONG. (FS)	C.G. LOCATION LAT. (BL)	DENSITY ALTITUDE (FT)	OAT (DEG. C)	ROTOR SPEED (RPM)	MODEL WEIGHT (FT)	CONFIGURATION
3340.	340.0(STA 3.0 @ 1)		2000.	8.6	390.	25.	1R

NOTES: 1. TRUE AIRSPEED IS VECTORIAL SUM OF WIND AND GROUND SPEED.
2. WINDS LESS THAN THREE KNOTS.
3. AFCS ON.
4. VERTICAL LINES DENOTE CONTROL AND AIRCRAFT EXCURSIONS.

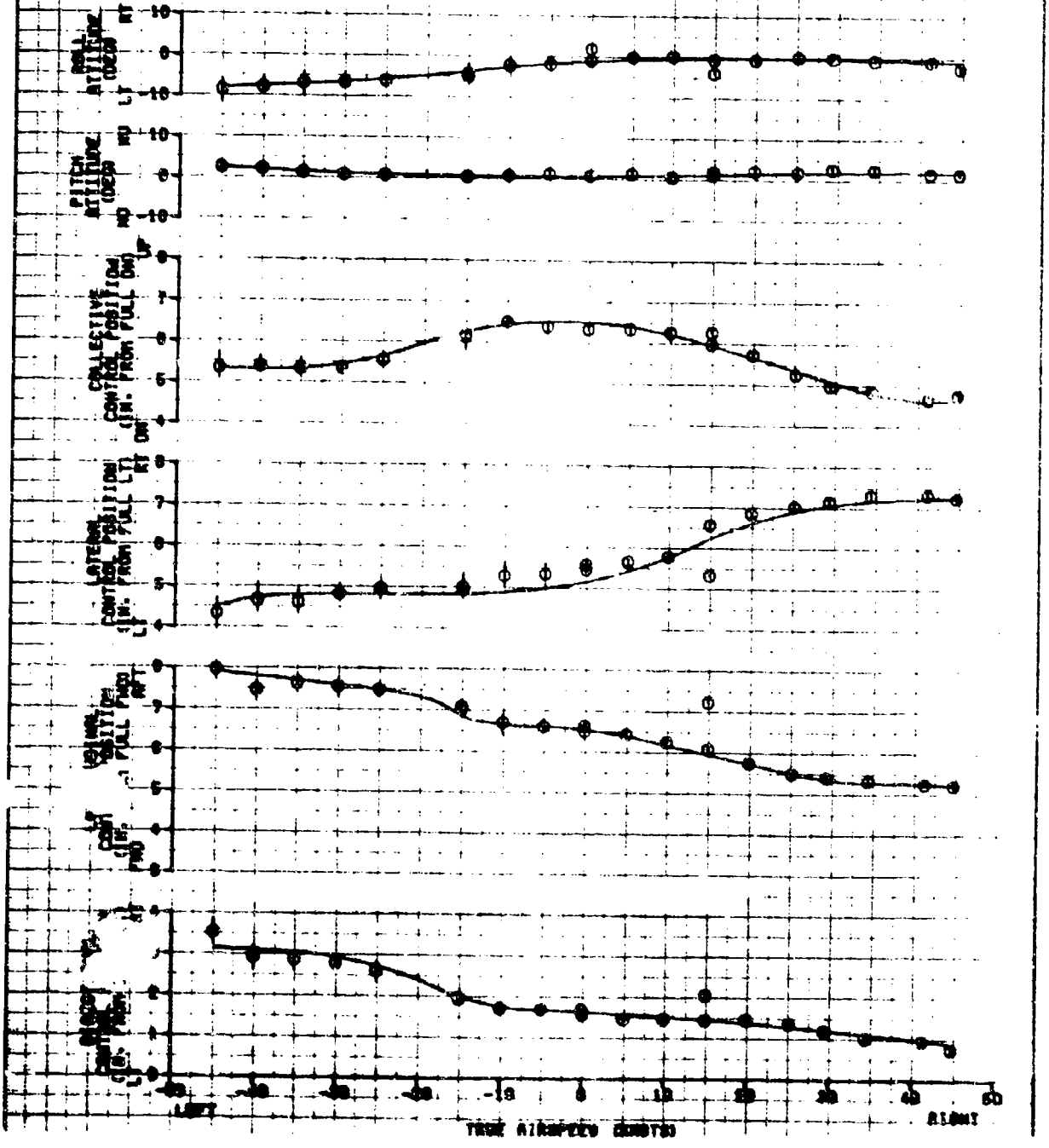


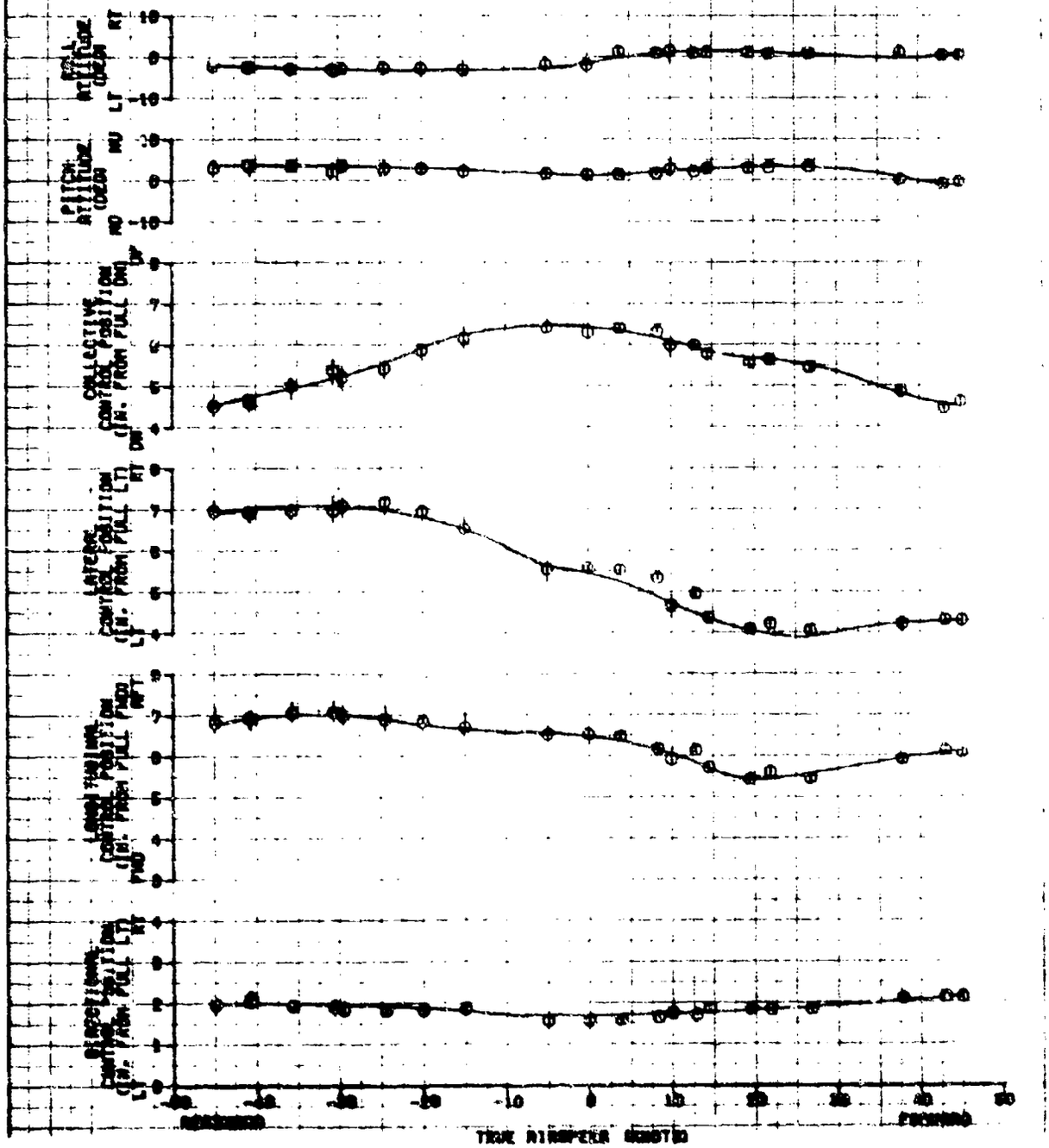
FIGURE 105

LOW SPEED FORWARD AND REARWARD FLIGHT CHARACTERISTICS

UH-60A S/N 77-21717

AVG GROSS WEIGHT (LBS)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG WHEEL HEIGHT (FT)	CONFIGURATION
16500	360 AFTER	0.3 (LT)	2220	7.8	258	25	1A

NOTES: 1. TRUE AIRSPEED IS VECTORIAL SUM OF WIND AND GROUND SPEED
2. WINDS LESS THAN THREE KNOTS
3. AFCS ON
4. VERTICAL LINES DENOTE CONTROL AND AIRCRAFT EXCURSIONS



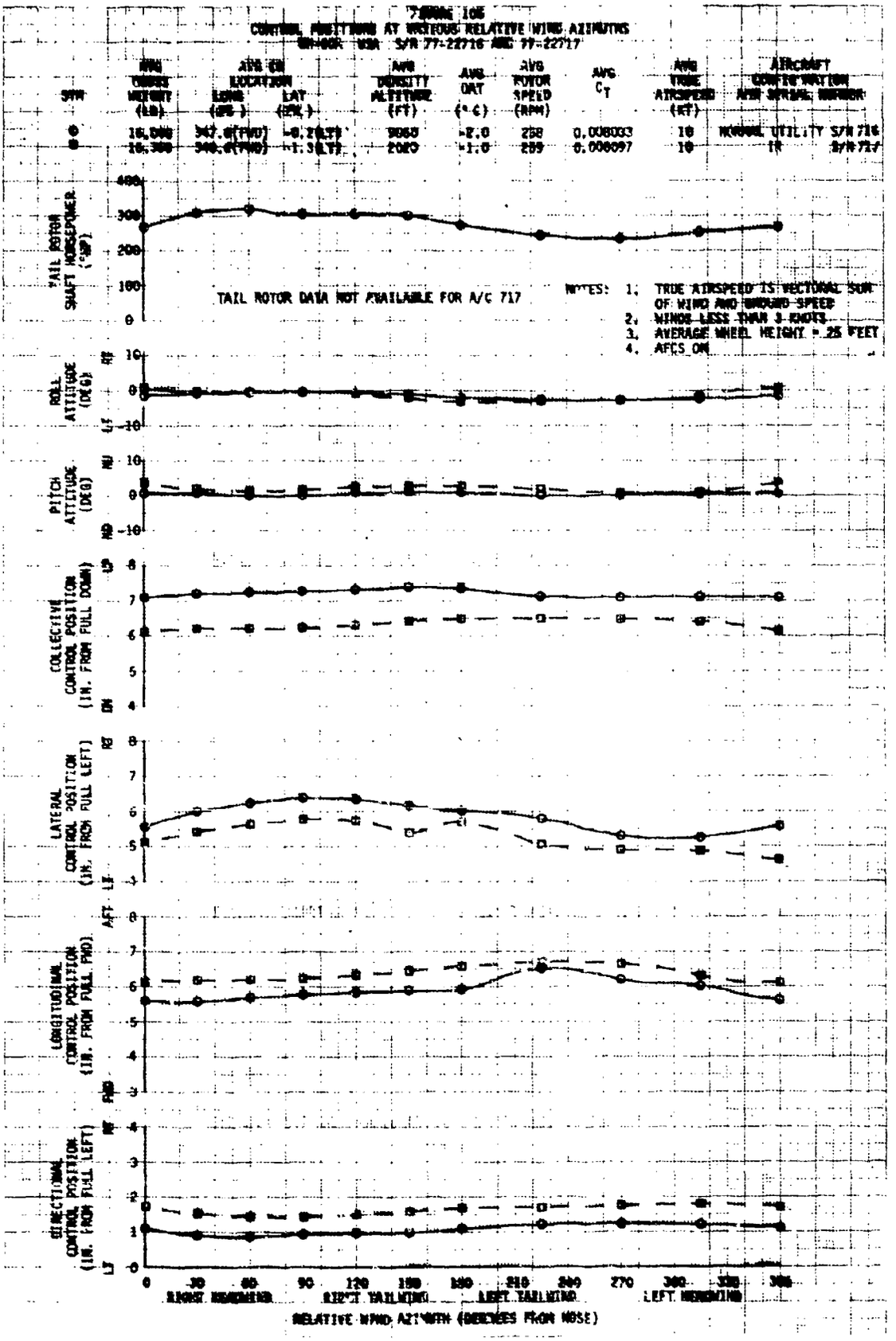

CONTROL POSITIONS AT VARIOUS RELATIVE WIND AZIMUTHS
S/N 77-22716 AND 77-22717
TAIL ROTOR DATA NOT AVAILABLE FOR A/C 717
NOTES: 1. TRUE AIRSPEED IS VECTORAL SUM OF WIND AND GROUND SPEED
2. WINDS LESS THAN 3 KNOTS
3. AVERAGE WHEEL HEIGHT = 25 FEET
4. AFCS ON
TAIL ROTOR SHAFT HORSEPOWER (SHP)
ROLL ATTITUDE (DEG)
PITCH ATTITUDE (DEG)
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
RELATIVE WIND AZIMUTH (DEGREES FROM NOSE)

FIGURE 187
CONTROL POSITIONS AT VARIOUS RELATIVE WIND AZIMUTHS
UH-60A USA S/N 77-22718 AND 79-22717

S/N	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	AVG TRUE AIRSPEED (KT)	AIRCRAFT CONFIGURATION AND SERIAL NUMBER
●	16,840	367.0(FWD)	-0.2(LT)	9660	-2.0	258	0.006833	20	NORMAL UTILITY S/N 718
○	16,360	366.6(FWD)	-1.3(LT)	9020	-1.0	259	0.006697	20	IR S/N 717

TAIL ROTOR DATA NOT AVAILABLE FOR A/C 717

NOTES:
1. TRUE AIRSPEED IS VECTORIAL SUM OF WIND AND GROUND SPEED
2. WINDS LESS THAN 3 KNOTS
3. AVERAGE WHEEL HEIGHT = 25 FEET
4. AFCS ON

TAIL ROTOR SHAFT HORSEPOWER (SHP)

ROLL ATTITUDE (DEG) RT / LT

PITCH ATTITUDE (DEG) NU / ND

COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN) UP / DN

LATERAL CONTROL POSITION (IN. FROM FULL LEFT) RT / LT

LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD) AFT / FWD

DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT) RT / LT

RIGHT HEADWIND | RIGHT TAILWIND | LEFT TAILWIND | LEFT HEADWIND

RELATIVE WIND AZIMUTH (DEGREES FROM NOSE)

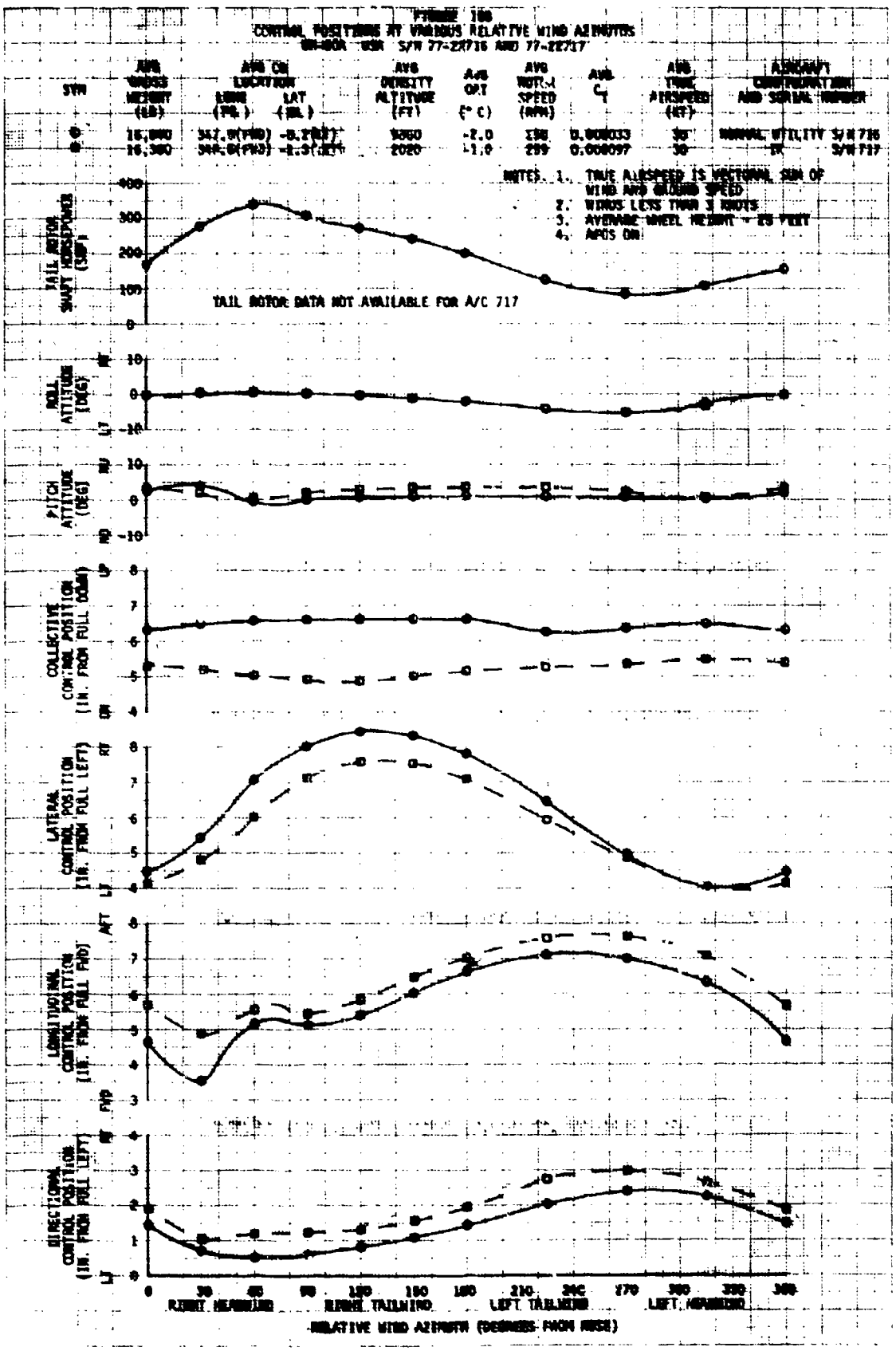
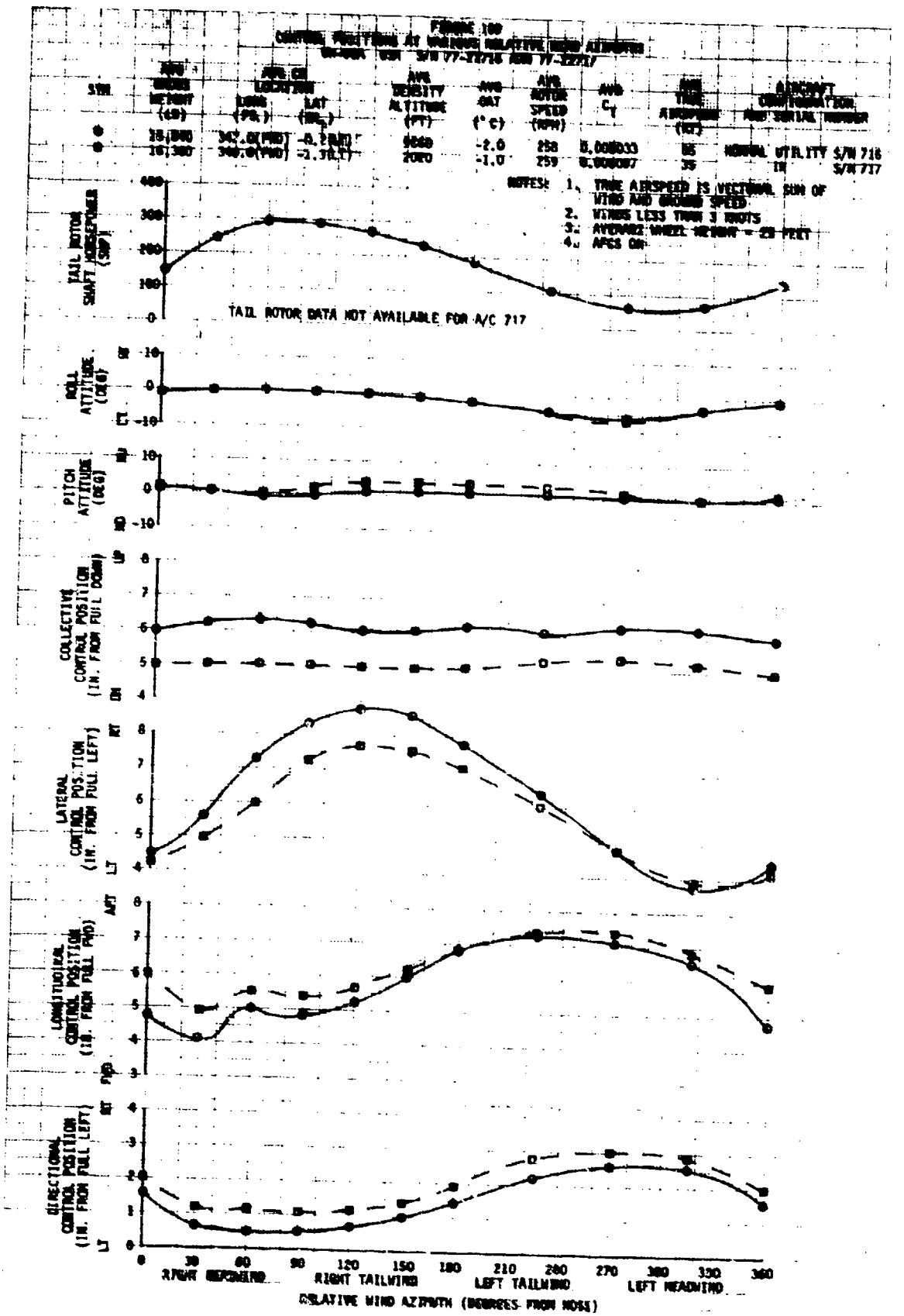


FIGURE 186
CONTROL POSITIONS AT VARIOUS RELATIVE WIND AZIMUTHS
S/N 77-22716 AND 77-22717
S/N
A/C ON LOCATION
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
AVG TRUE AIRSPEED (KT)
AIRCRAFT CONFIGURATION AND SERIAL NUMBER
16,000
16,300
2020
-2.0
-1.0
239
30
NAVAL UTILITY S/N 716
S/N 717
NOTES: 1. TRUE AIRSPEED IS VECTORIAL SUM OF WIND AND GROUND SPEED
2. WINDS LESS THAN 3 KNOTS
3. AVERAGE WHEEL HEIGHT = 25 FEET
4. AFDS ON
TAIL ROTOR DATA NOT AVAILABLE FOR A/C 717
TAIL ROTOR SHAFT HORSEPOWER (SHP)
ROLL ATTITUDE (DEG)
PITCH ATTITUDE (DEG)
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
RELATIVE WIND AZIMUTH (DEGREES FROM NOSE)


TAIL ROTOR DATA NOT AVAILABLE FOR A/C 717
TAIL ROTOR SHAFT HORSEPOWER (SHP)
ROLL ATTITUDE (DEG)
PITCH ATTITUDE (DEG)
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
RIGHT HEADWIND
RIGHT TAILWIND
LEFT TAILWIND
LEFT HEADWIND
RELATIVE WIND AZIMUTH (DEGREES FROM NOSE)

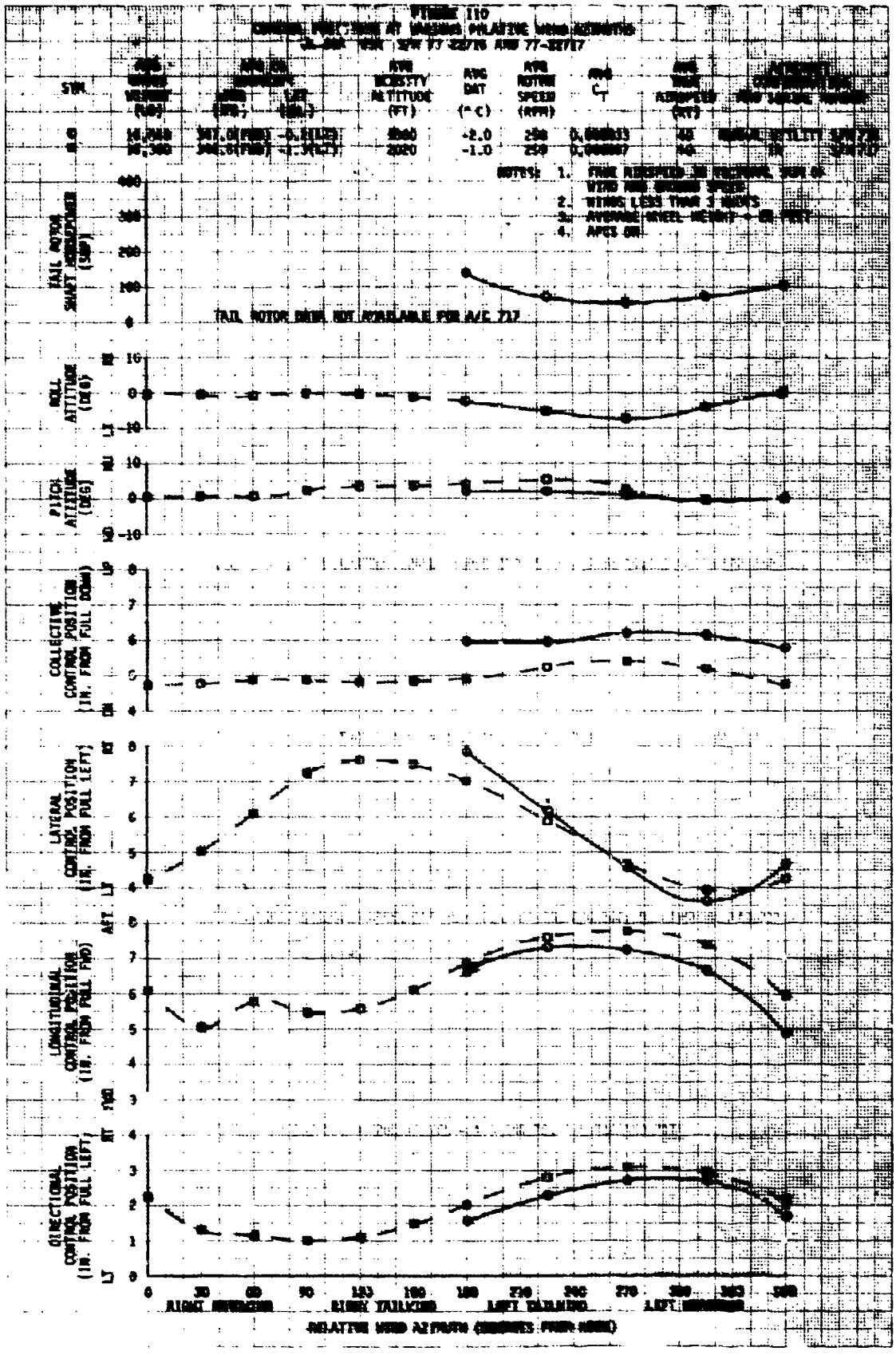


FIGURE 110
STR
AVG WIND VELOCITY (KNS)
AVG WIND DIRECTION (DEG, TRUE)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
AVG C_T
AVG TRUE AIRSPEED (KT)
ACCEPTABLE CONTROL POSITION AND VARIANCE TOLERANCE
-0.0
-1.0
250
NOTES: 1.
2.
3.
4. APCS ON
TAIL ROTOR DATA NOT AVAILABLE FOR A/C 717
TAIL ROTOR SHAFT HORSEPOWER (SHP)
ROLL ATTITUDE (DEG)
PITCH ATTITUDE (DEG)
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
RIGHT WINDOW
RIGHT TAILSIDE
LEFT TAILSIDE
LEFT WINDOW
RELATIVE WIND AZIMUTH (DEGREES FROM NOSE)

FIGURE 111

SUCCESSFUL LEFT ENGINE RESTART USING APU

UH-60A USA S/N 77-22717

SOLID LINE	SHORT DASH	LONG DASH	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KTS)	CONFIGURATION
			13640	363.1 (AFT)	-1.2 (LT)	18700	-11.5	258	90	NORMAL IDLE

RIGHT ENGINE

TORQUE (FT-LB) 500 400 300 200 100 0

MEASURED GAS TEMPERATURE (DEG C) 1000 800 600 400 200 0

FUEL FLOW RATE (GAL/HR) 100 80 60 40 20 0

GAS GENERATOR SPEED (PERCENT) 100 80 60 40 20 0

TORQUE INDICATING SYSTEM FLUCTUATIONS

COLLECTIVE CONTROL POSITION (IN. FROM FULL DN) 10 8 6 4 2 0

LEFT ENGINE

TORQUE (FT-LB) 500 400 300 200 100 0

MEASURED GAS TEMPERATURE (DEG C) 1000 800 600 400 200 0

MAIN ROTOR SPEED (RPM) 270 265 260 255 250 245

FUEL FLOW RATE (GAL/HR) 100 80 60 40 20 0

GAS GENERATOR SPEED (PERCENT) 100 80 60 40 20 0

0 8 16 24 32 40 48 55 64 72 80 88 96 104 112 120

TIME - SECONDS

FIGURE 112

UNSUCCESSFUL LEFT ENGINE START USING AFU

UH-60A USA S/N 77-22717

SOLID LINE	SHORT DASH	LONG DASH	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KTS)	CONFIGURATION
			13500	364.0 (AFT)	-1.1 (LT)	21080	-23.5	258	40	NORMAL UTIL

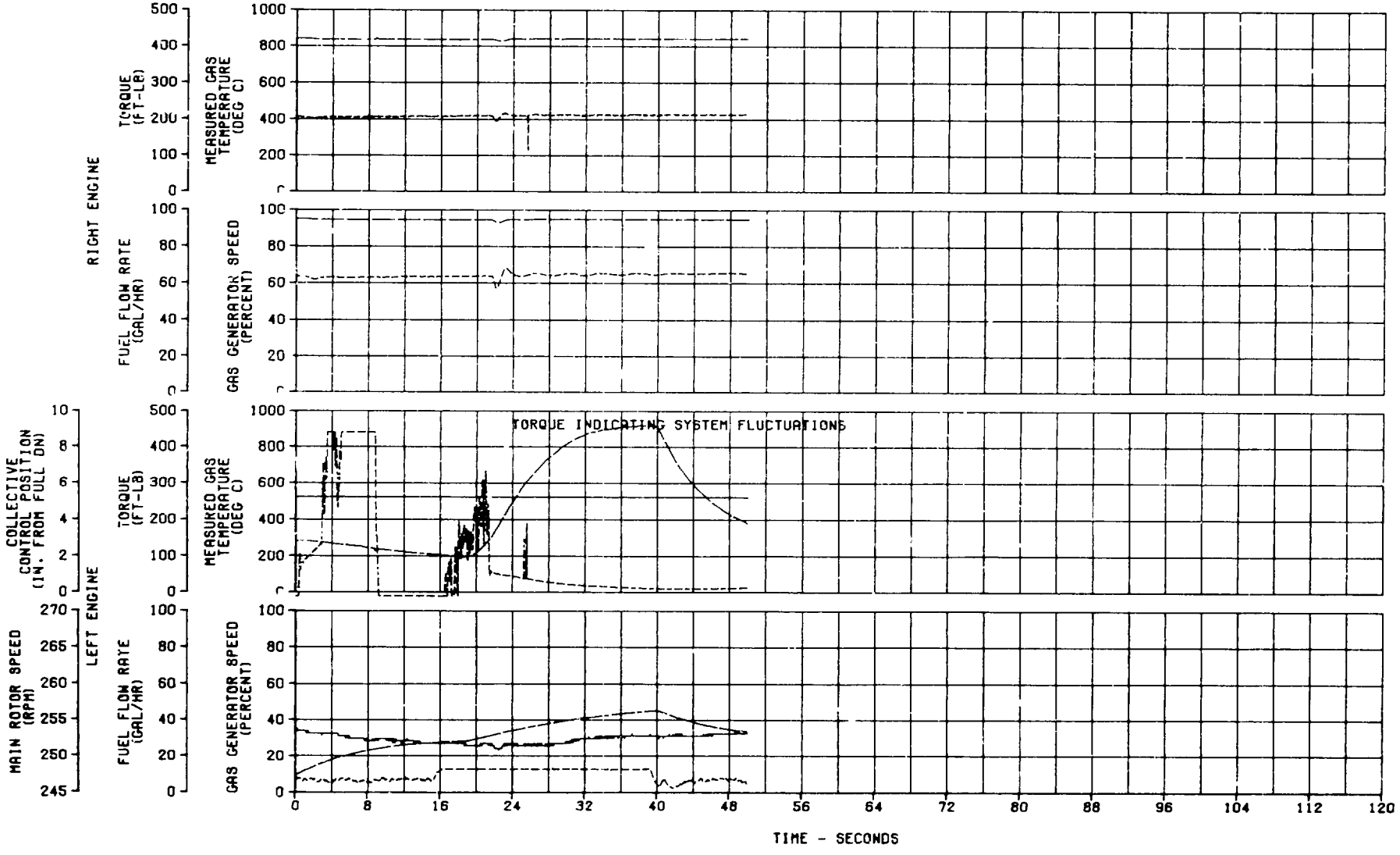


FIGURE 113
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
PILOT STATION
4/REV (17.2 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
16,360	349.1 (EWD)	0.0 (MID)	6180	12.0	257	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION

INSTRUMENTATION INOPERATIVE

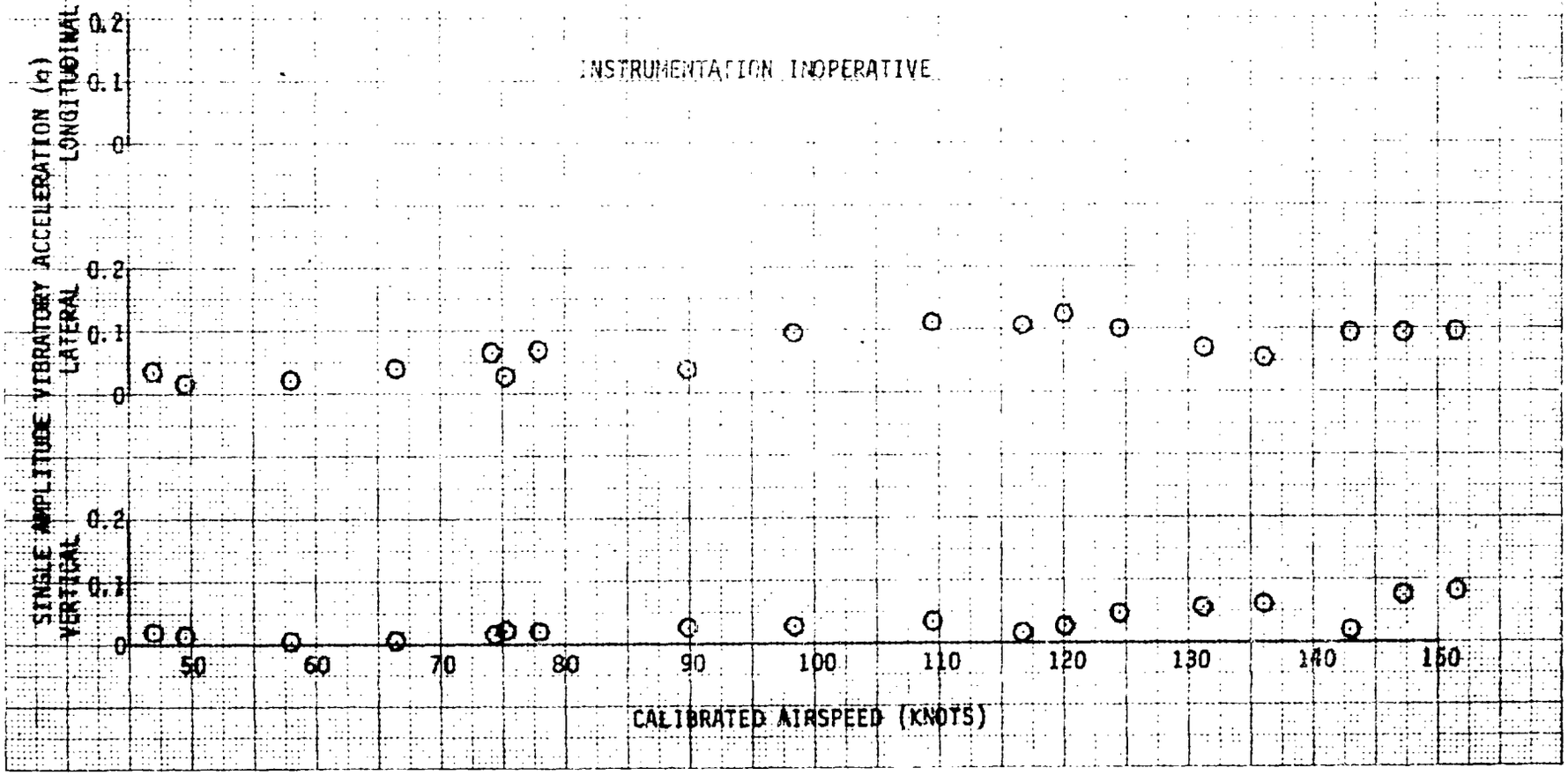


FIGURE 114
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
PILOT STATION
4/REV (17.2 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
16,480	347.1(FWD)	0.0(MID)	6400	-19.6	258	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION

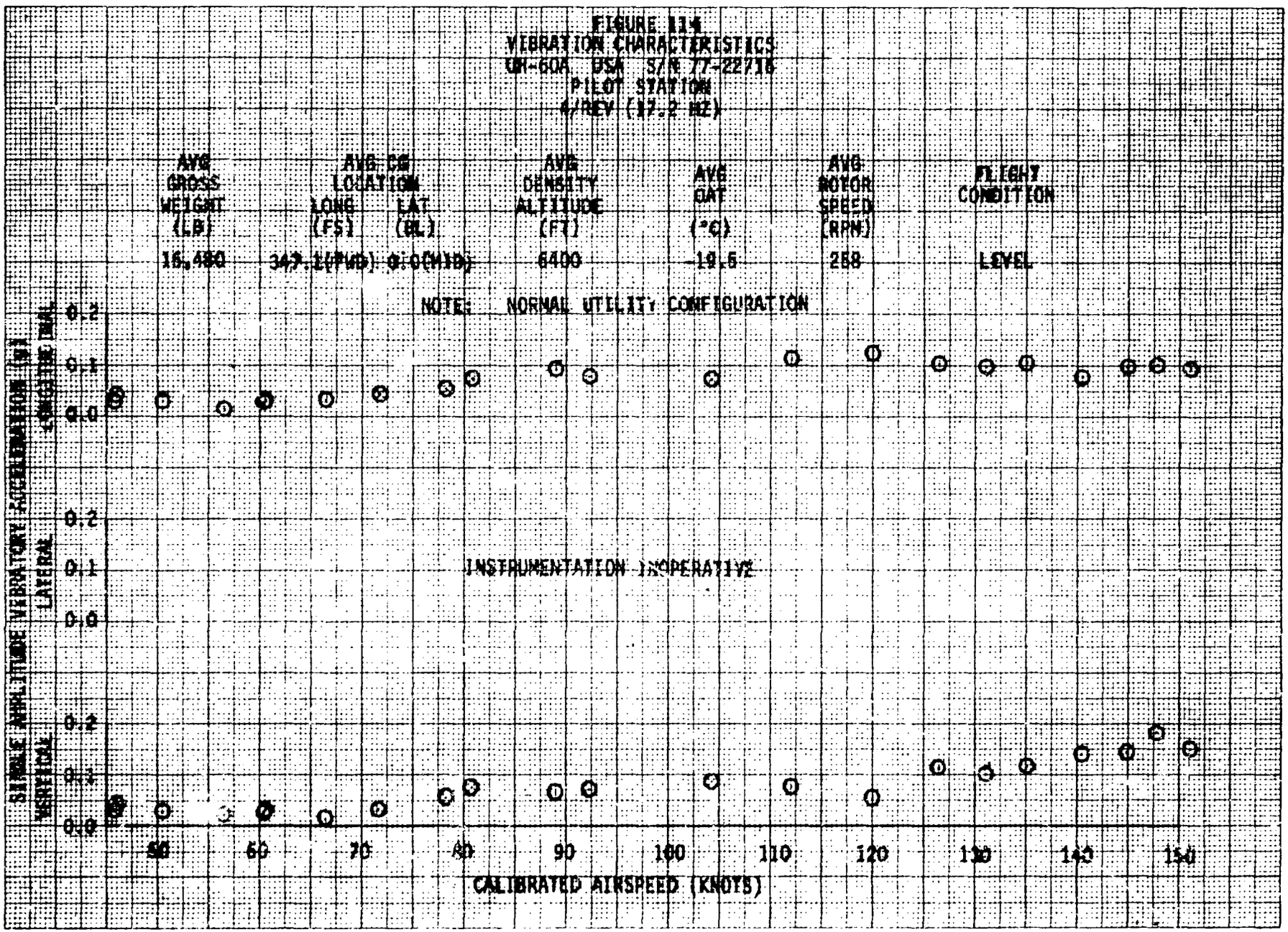
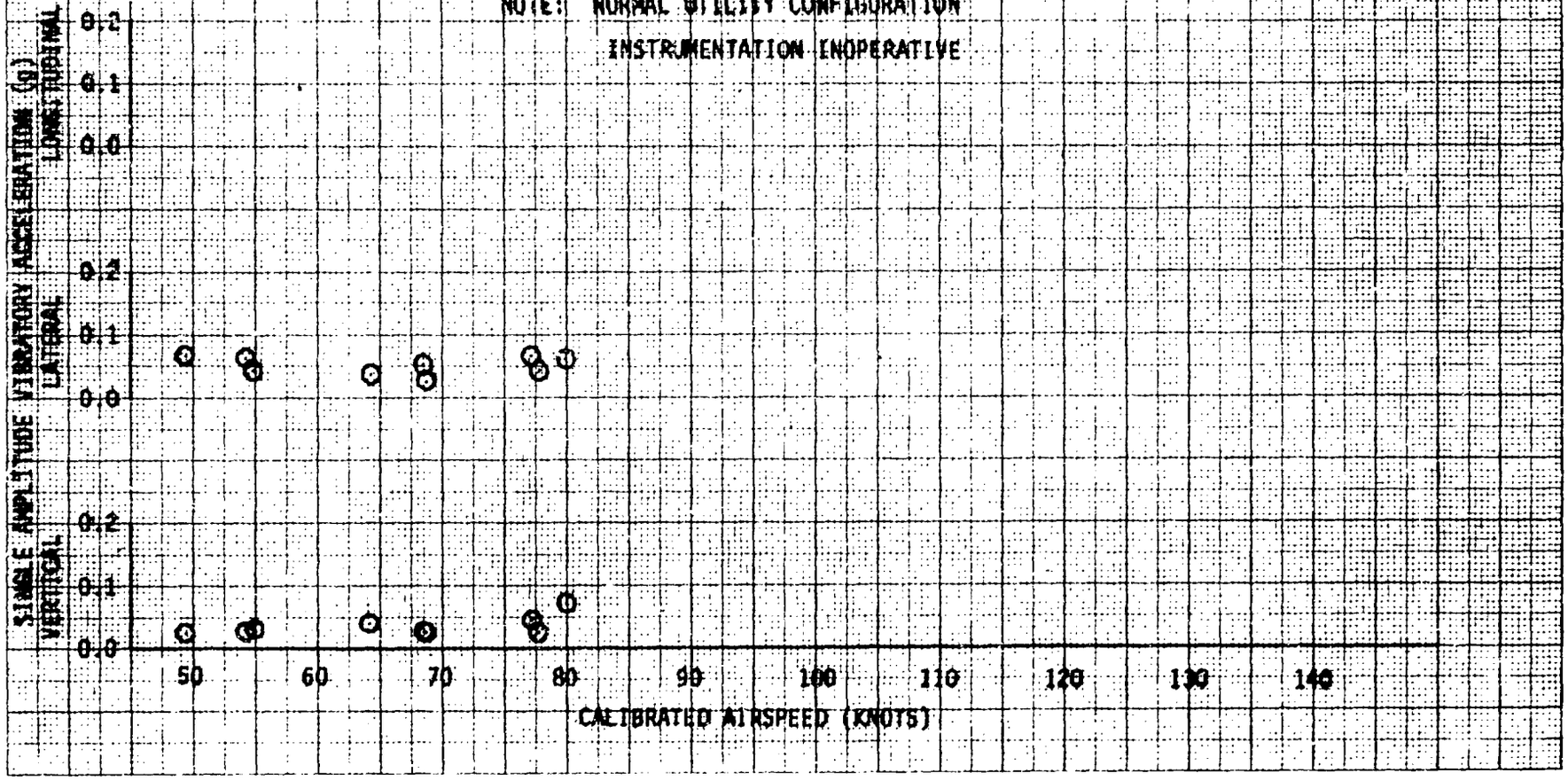


FIGURE 116
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
PILOT STATION
4/REV (16.8 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPN)	FLIGHT CONDITION
18,880	347.2(FWD)	0.0(MID)	14,740	-32.0	252	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION
INSTRUMENTATION INOPERATIVE



CALIBRATED AIRSPEED (KNOTS)
SINGLE MAIN ROTOR HELICOPTER MODE CONTROL (δ)
LATERAL
LONGITUDINAL
NORMAL UTILITY CONFIGURATION
NOTE:
AVG GROSS WEIGHT
AVG DENSITY ALTITUDE
AVG ROTOR SPEED
LEVEL

FIGURE 117
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
CENTER OF GRAVITY
4/REV (17.2 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
16,480	347.1(FWD)	0.0(MID)	6400	-19.5	258	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION

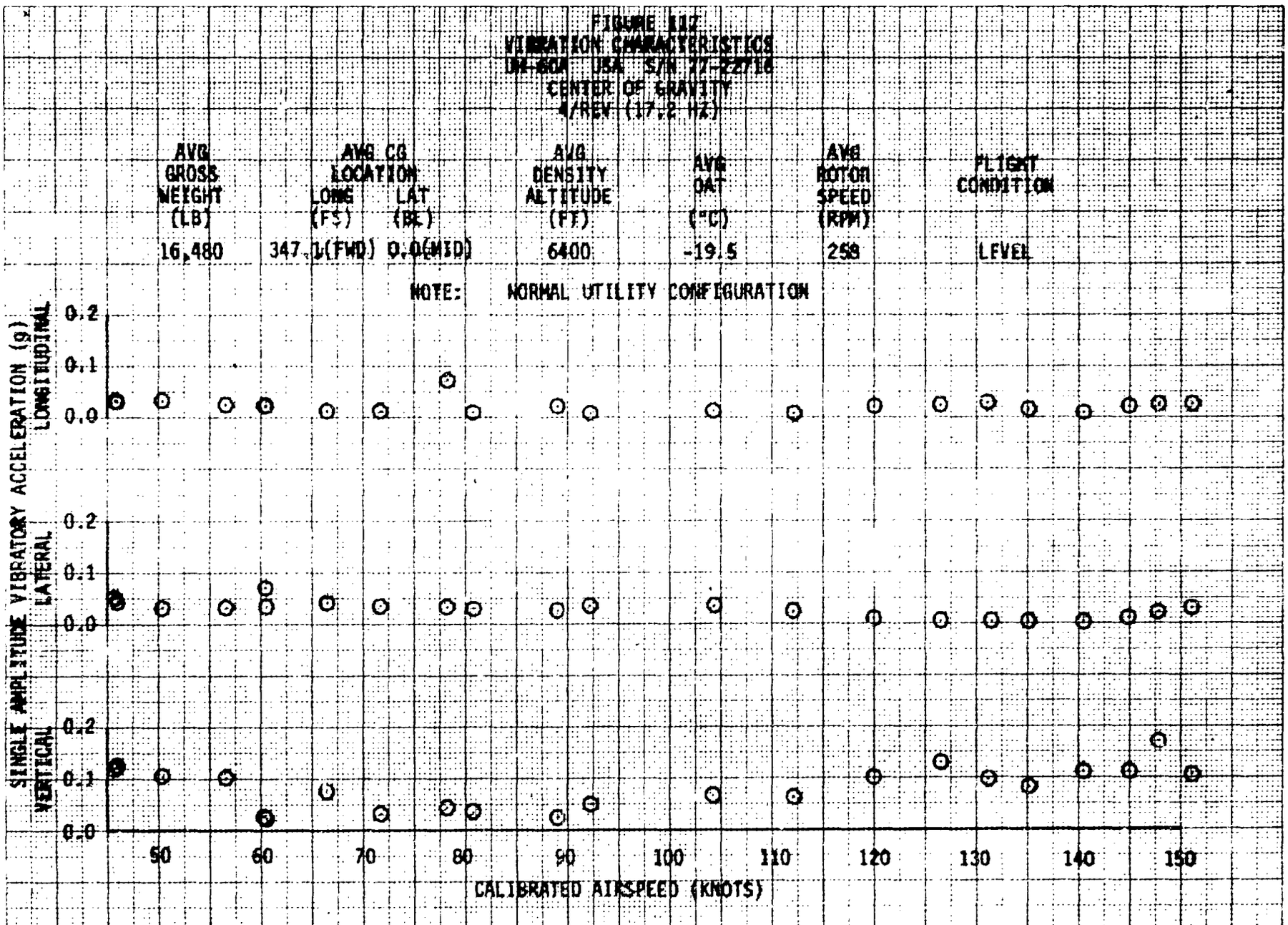


FIGURE 116
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
CENTER OF GRAVITY
4/REV (16.8 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
18,880	347.2(FWD)	0.0(MID)	14,740	-32.0	252	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION

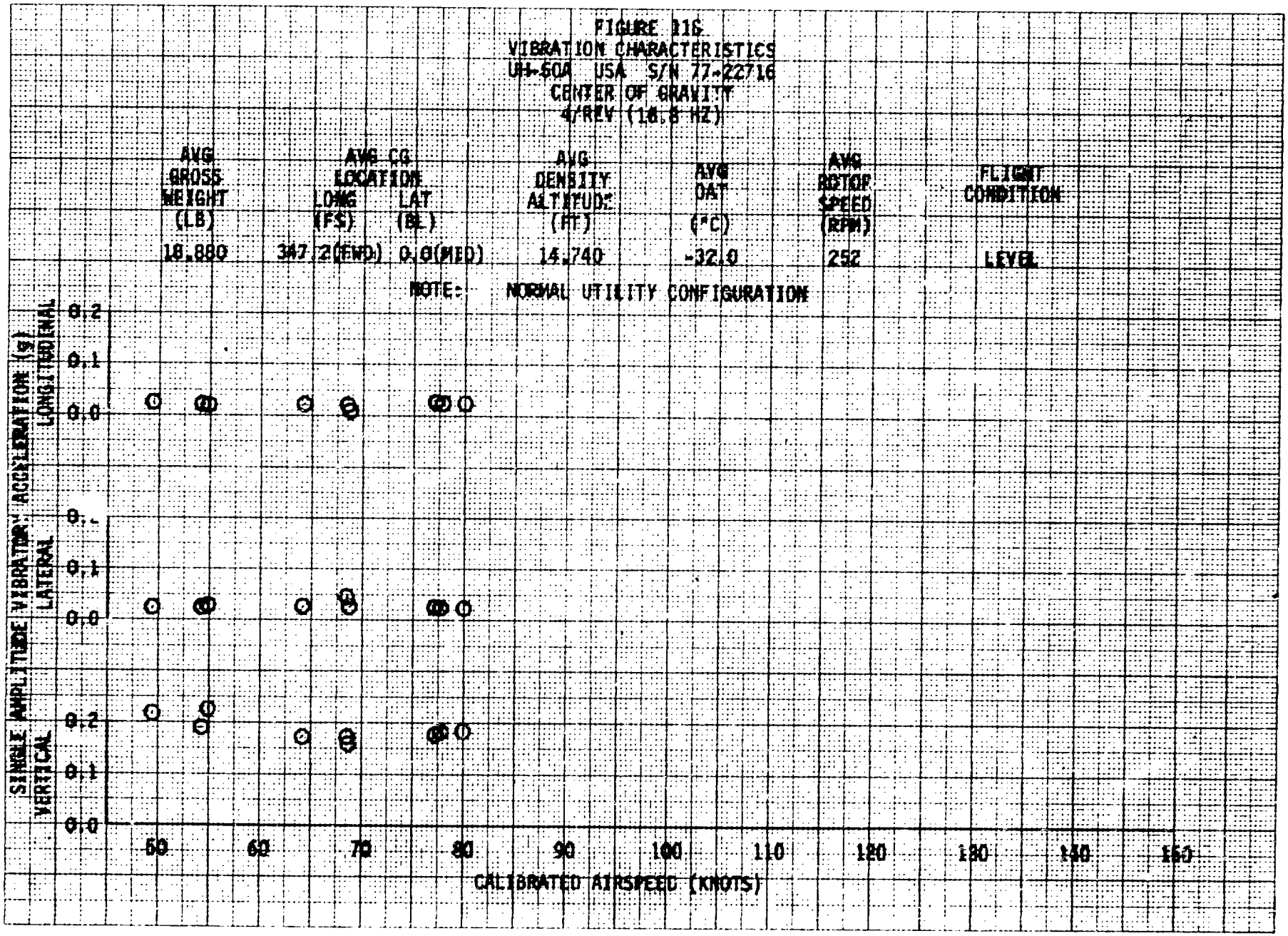


FIGURE 119
VIBRATION CHARACTERISTICS
OH-6CA USA S/N 77-22716
PILOT STATION
4/REV (16.3 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
16,660	347.2(FWD)	0.06(RIG)	6880	14.5	245	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION

INSTRUMENTATION INOPERATIVE

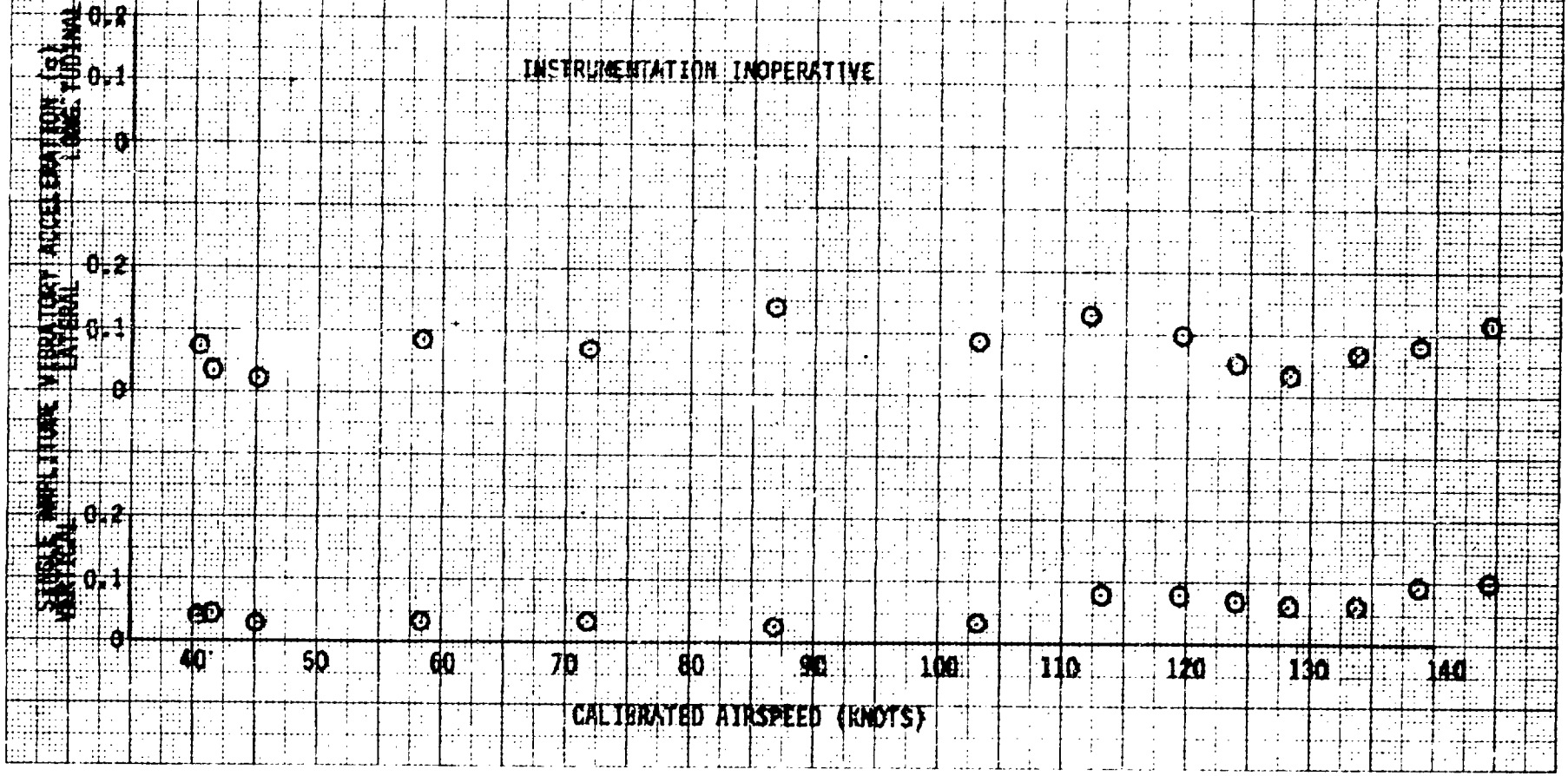
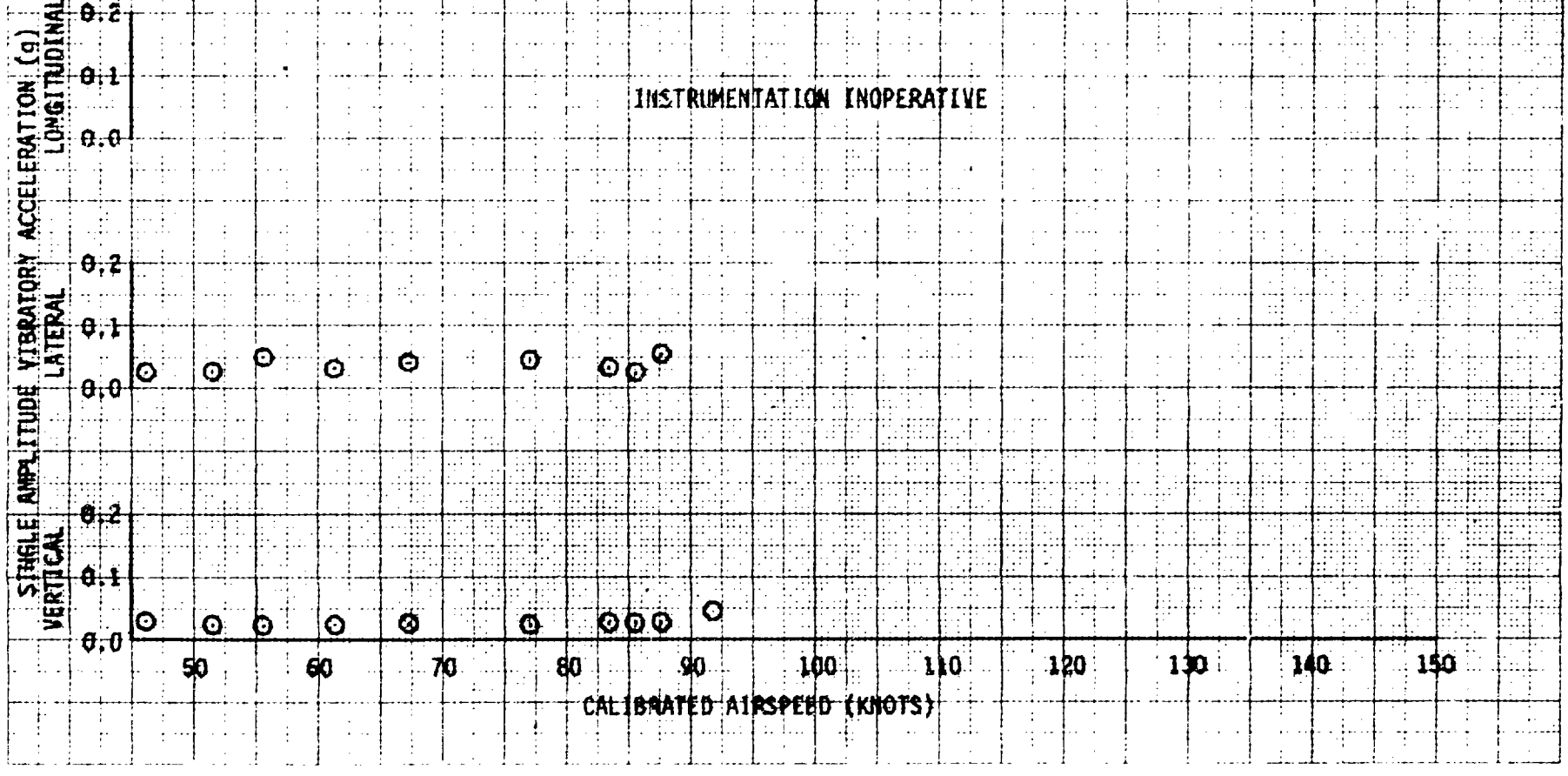


FIGURE 12B
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
PILOT STATION
4/REV (16.4 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
19,780	347.0(FWD)	0.0(MID)	11,920	-11.5	246	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION



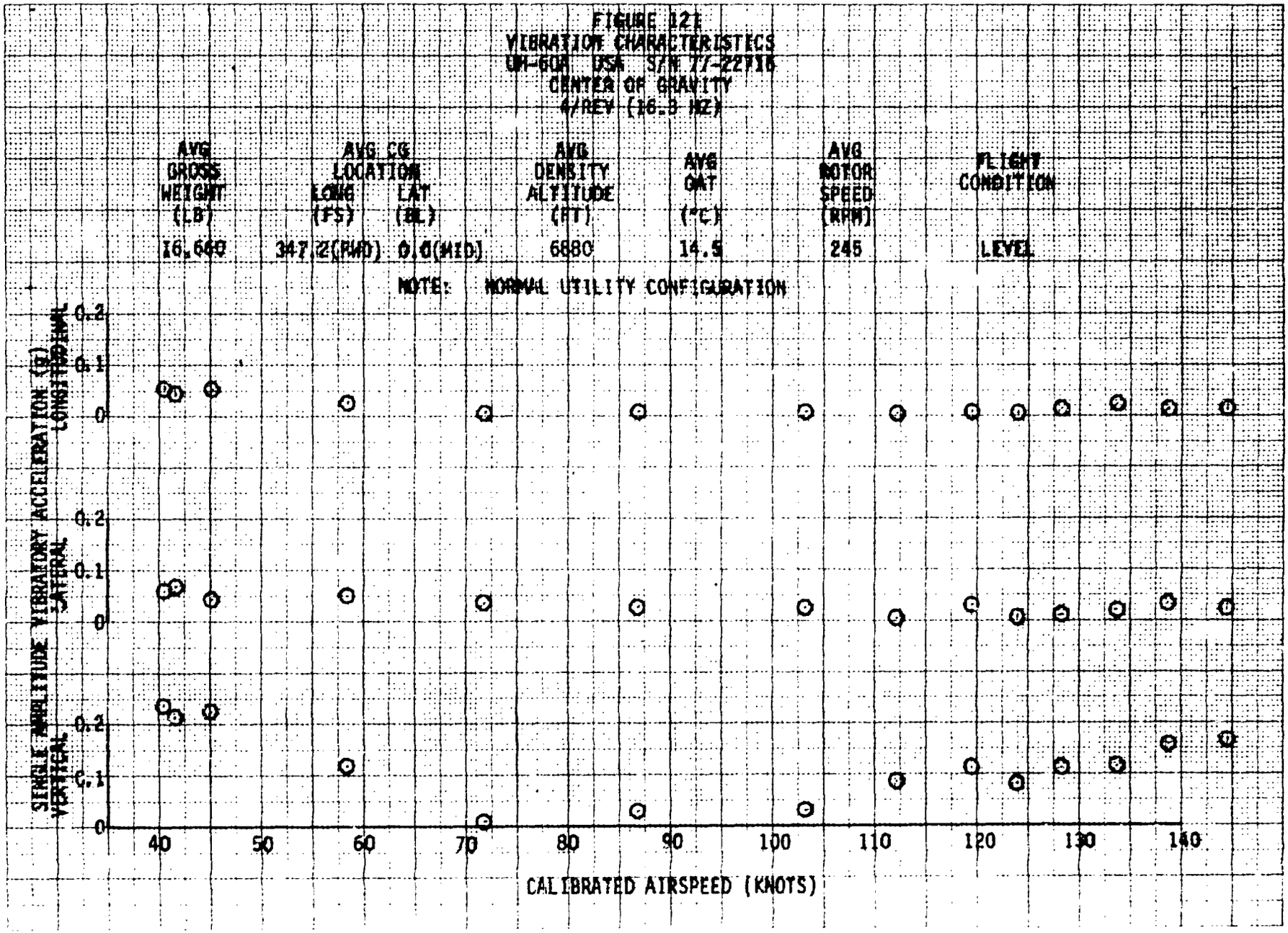


FIGURE 121
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
CENTER OF GRAVITY
4/REV (16.8 HZ)
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
16,660
347.2(FWD)
0.0(MID)
6880
14.5
245
LEVEL
NOTE: NORMAL UTILITY CONFIGURATION
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
VERTICAL
LONGITUDINAL
LATERAL
0.2
0.1
0
40
50
60
70
80
90
100
110
120
130
140
CALIBRATED AIRSPEED (KNOTS)

FIGURE 122
VIBRATION CHARACTERISTICS
UH-60A USA S/N 77-22716
CENTER OF GRAVITY
4/REV (16.4 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
19,780	347.0(FWD)	0.0(MID)	11,920	-11.5	246	LEVEL

NOTE: NORMAL UTILITY CONFIGURATION

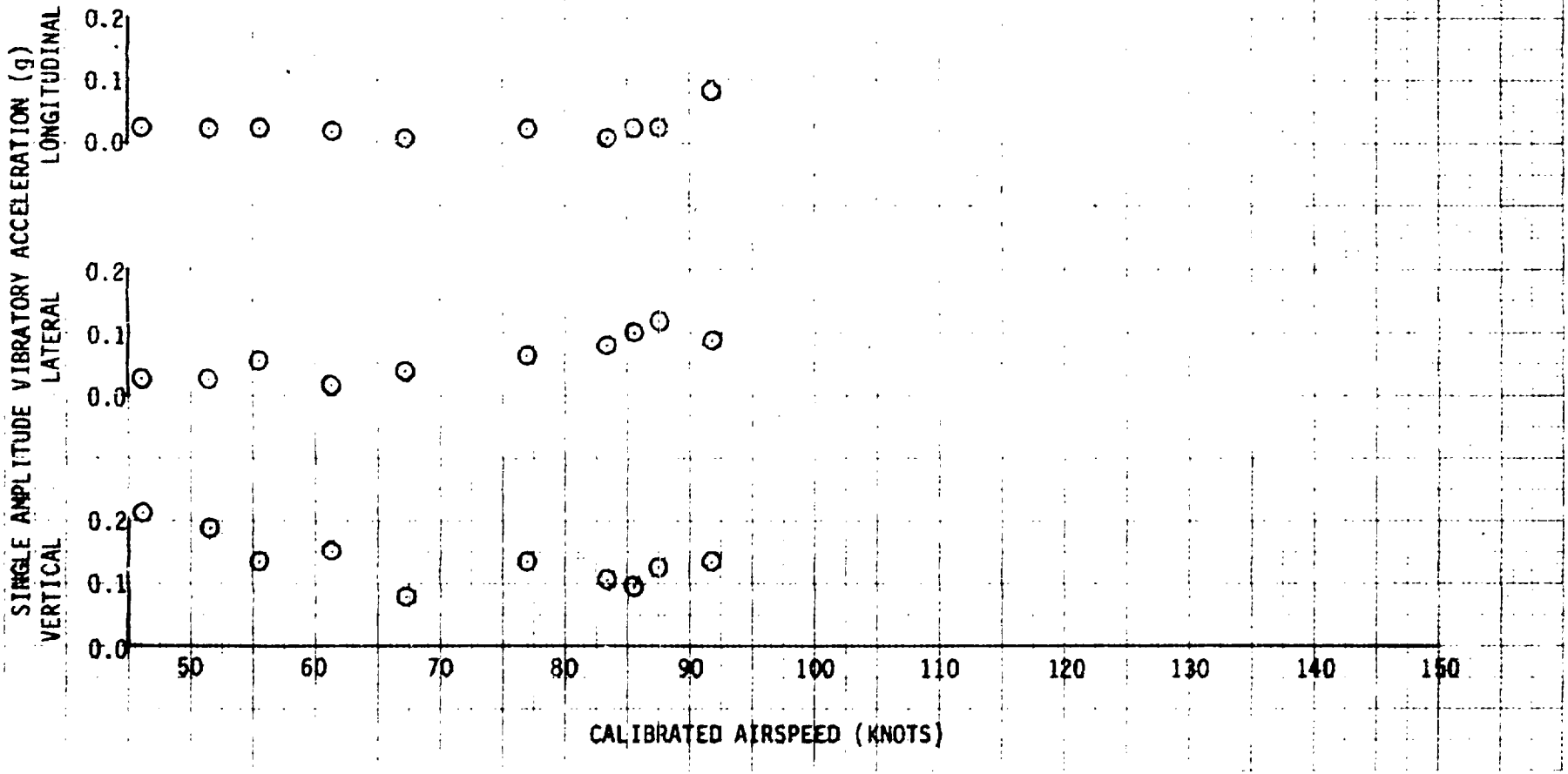


FIGURE 123.
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
UH-60A USAF S/N 77-22716
PILOT STATION
4/REV (17.2 HZ)

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
△	16,720	359.0(AFT)	0.2(RT)	6520	11.0	258	116	LT TURN
○	16,740	359.0(AFT)	0.2(RT)	6500	11.0	258	116	RT TURN

NOTE: NORMAL UTILITY CONFIGURATION

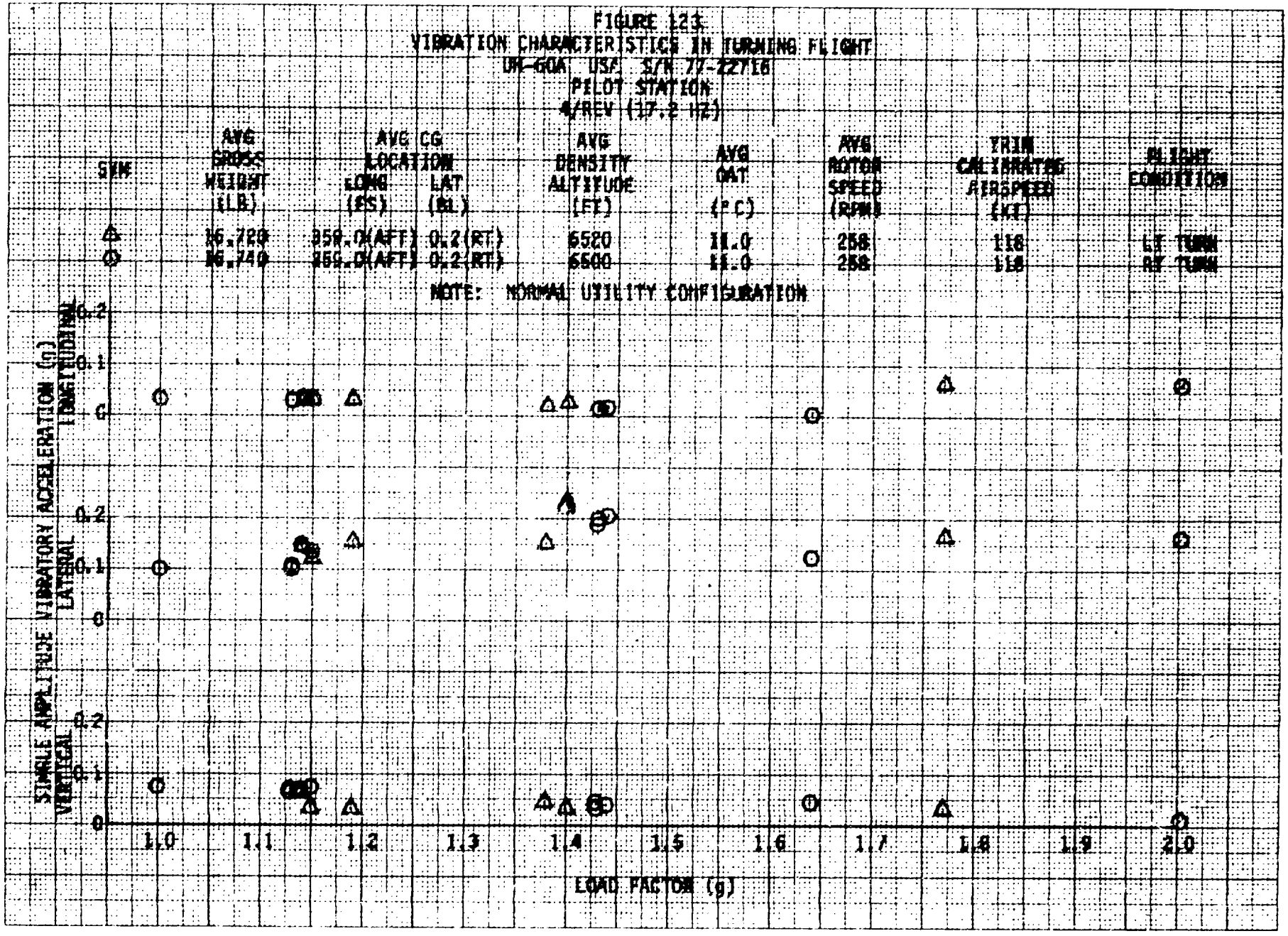


FIGURE 124
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
UH-60A USA S/N 77-22716
CENTER OF GRAVITY
4/REV (17.2 HZ)

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
△	16,720	359.0(AFT)	0.2(RT)	6520	11.0	258	118	LH TURN
○	16,740	359.0(AFT)	0.2(RT)	6500	11.0	258	118	RT TURN

NOTE: NORMAL UTILITY CONFIGURATION

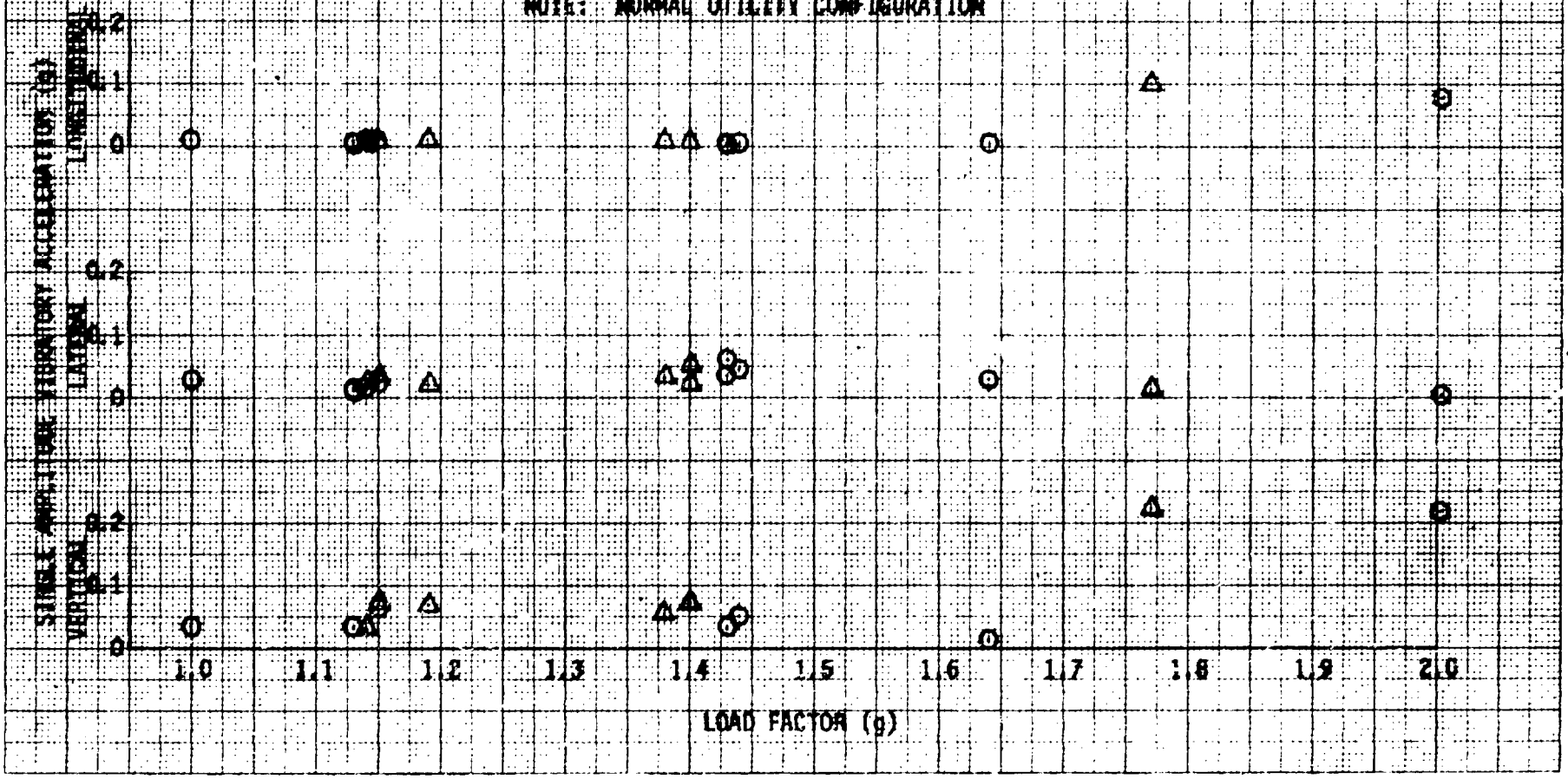


FIGURE 125

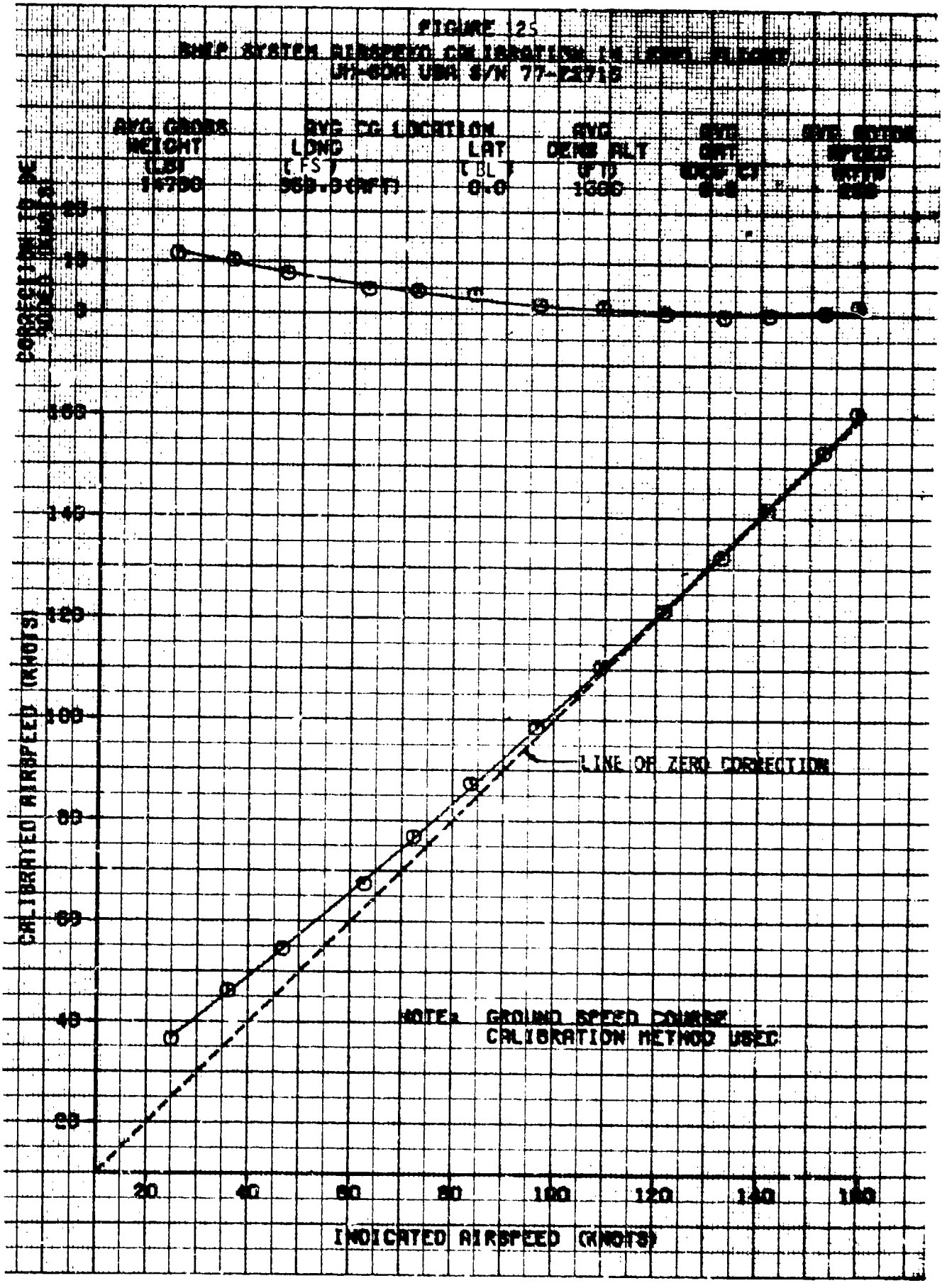
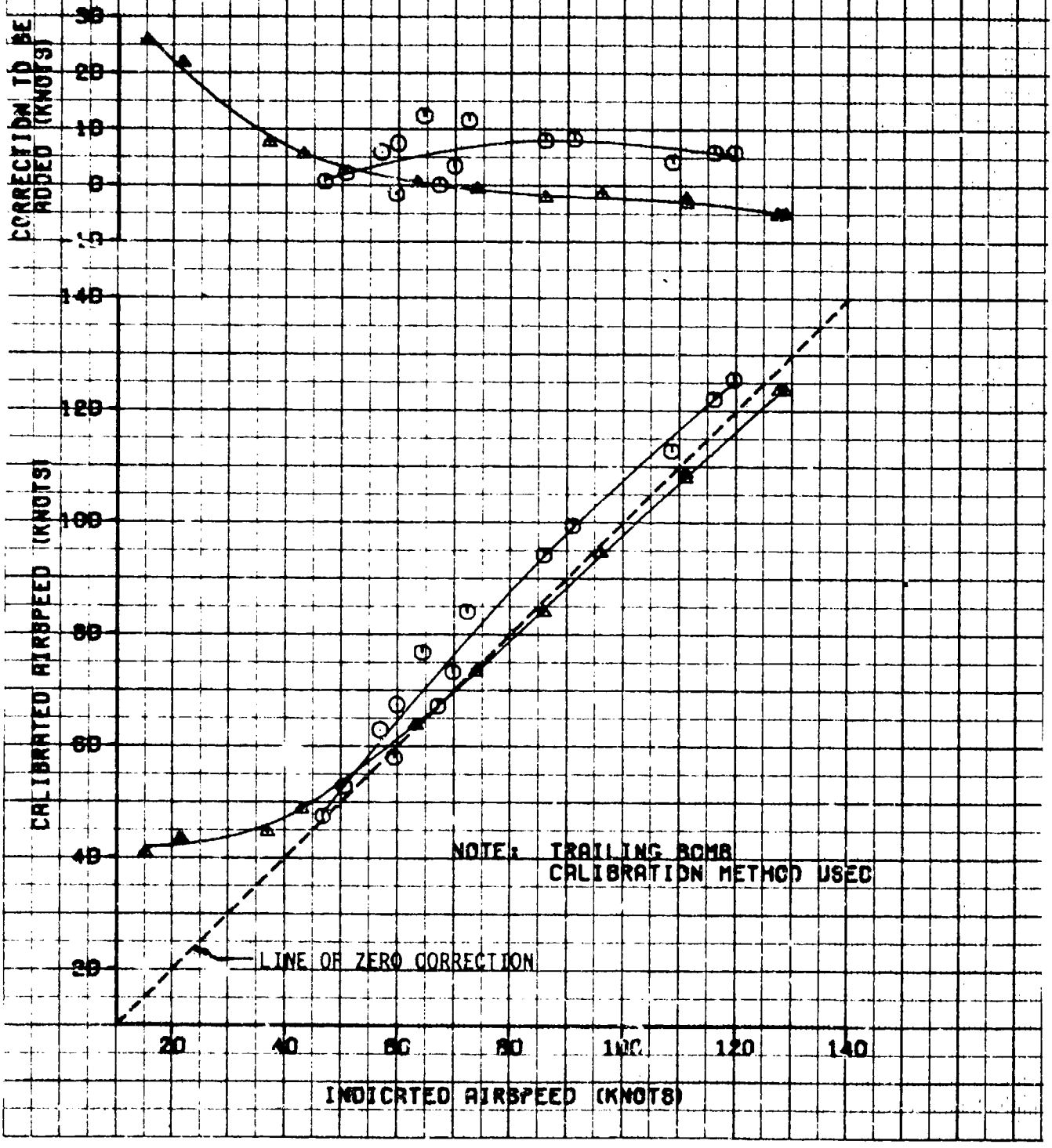

AVG GROSS WEIGHT (LBS) 14700
AVG CG LOCATION LONG (FS) 369.0 (AFT)
LAT (BL) 0.0
AVG DENS ALT (FT) 1000
CALIBRATED AIRSPEED (KNOTS)
INDICATED AIRSPEED (KNOTS)
LINE OF ZERO CORRECTION
NOTE: GROUND SPEED COURSE CALIBRATION METHOD USED

FIGURE 126

SHIP SYSTEM AIRSPEED CALIBRATION IN CLIMBING AND AUTOROTATIONAL FLIGHT
UH-60A USA S/N 77-22718

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENS ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLT COND
⊙	16900	347.3 (FWD)	0.0	7800	14.0	258	CLIMB
▲	16900	347.3 (FWD)	0.0	7500	13.0	258	AUTO



APPENDIX F. GLOSSARY

A&FC	airworthiness and flight characteristics
AC	alternating current
ADF	automatic direction finder
AFCS	automatic flight control system
AGL	above ground level
APU	auxiliary power unit
AVRADCOM	US Army Aviation Research and Development Command
BL	butt line
CB	crossbleed
cg	center of gravity
CIS	command instrument system
C_T	Coefficient of thrust
DC	direct current
ECU	engine electrical control unit
EMI	electromagnetic interference
FAT	free air temperature
F_e	equivalent flat plate area
FM	frequency modulator
fpm, ft/min	feet per minute
FPS	flight path stability
FS	fuselage station
g	acceleration of gravity
GCT	Government Competitive Test
GE	General Electric
guard	to remain close to a flight control without touching it
HMU	engine hydromechanical unit
HQRS	Handling Qualities Rating Scale
HSI	horizontal situation indicator
Hz	Hertz
H_p	pressure altitude
ICP	intercommunication control panel
IGE	in ground effect
ILS	instrument landing system
IMC	instrument meteorological conditions
IR	infrared
IRP	intermediate rated power
KIAS	knots indicated airspeed
KCAS	knots calibrated airspeed
KTAS	knots true airspeed
K_P	power correction factor
K_W	weight correction factor
MCP	maximum continuous power
NAV	navigation
N_G	engine gas generator speed
N_P	engine power turbine speed
N_R	rotor speed
OAT	outside air temperature
OBS	omni-bearing selector
OGE	out of ground effect
PAE	preliminary airworthiness evaluation
PBA	pitch bias actuator
PCL	power control lever
PIDS	Prime Item Development Specification
PVT-G	Production Validation Test-Government

RAM	reliability, availability, and maintainability
RPM	revolutions per minute
SAS	stability augmentation system
SHP	shaft horsepower
TEDN	trailing edge down
TEUP	trailing edge up
TGT	turbine gas temperature
USAAEFA	United States Army Aviation Engineering Flight Activity
VOR	very high frequency omni-directional range
VRS	Vibrations Rating Scale
VSI	vertical situation indicator
V_H	maximum attainable airspeed in level flight at intermediate rated power
$V_{min\ R/D}$	airspeed for minimum rate of descent
Δ	incremental change
μ	main rotor advance ratio
°C	temperature degrees centigrade
4/rev	fourth harmonic of the main rotor
$N_R/\sqrt{\theta}$	referred rotor speed
w/δ	ratio of gross weight to pressure ratio

APPENDIX G. EQUIPMENT PERFORMANCE REPORT

The following Equipment Performance Reports (EPR's) were submitted during the Model UH-60A A&FC program.

EPR No.	Date Submitted	Descriptive Title
77-17-1	18 Sep 79	Auxiliary power unit misalignment of attaching bolt holes.
77-17-2	20 Nov 79	Faulty BIM indicator.
77-17-3	20 Nov 79	Auxiliary power unit heat shield could not be alligned properly.
77-17-4	28 Dec 79	Faulty hydraulic pump module.
77-17-5	28 Dec 79	Tail rotor gear box failed acidulated copper-sulphate solution test.
77-17-6	28 Dec 79	Faulty main rotor blade.
77-17-7	7 Jan 80	Faulty bifilar washer.
77-17-8	15 Jan 80	Chaffed first stage tail rotor hydraulic return line.
77-17-9	16 Jan 80	Broken bonding jumper.
77-17-10	27 Feb 80	Cracked left exhaust fairing.
77-17-11	18 Mar 80	Worn boot on damper.
77-17-12	18 Mar 80	Faulty main rotor spindle elastomeric bearing.
77-17-13	30 Apr 80	Damaged spindles and damper bearings.
77-17-14	30 Sep 80	Scorched inlet for the IR suppressor.
77-17-15	24 Oct 80	Cracked elastomeric bearing on the engine gimbal.
77-17-16	9 Sep 81	Engine shutdown when power lever was pulled aft of the detent position.

DISTRIBUTION

Deputy Chief of Staff for Logistics (DALO-SMM, DALO-AV)	2
Deputy Chief of Staff Operations (DAMO-RQ)	1
Deputy Chief of Staff for Personnel (DAPE-HRS)	1
Deputy Chief of Staff for Research Development and Acquisition (DAMA-PPM-T, DAMA-RA, DAMA-WSA)	3
Comptroller of the Army (DACA-ZA)	1
US Army Materiel Development and Readiness Command (DRCDE-SA, DRCQA-E, DRCDE-I, DRCDE-P)	4
US Army Training and Doctrine Command (ATTG-U, ATCD-T, ATCD-ET, ATCD-B)	4
US Army Aviation Research and Development Command (DRDAV-DI, DRDAV-EE, DRDAV-EG)	10
US Army Test and Evaluation Command (DRSTE-CT-A, DRSTE-TO-O)	2
US Army Troop Support and Aviation Materiel Readiness Command (DRSTS-Q)	1
US Army Logistics Evaluation Agency (DALO-LEI)	1
US Army Materiel Systems Analysis Agency (DRXSY-R, DRXSY-MP)	2
US Army Operational Test and Evaluation Agency (CSTE-POD)	1
US Army Armor Center (ATZK-CD-TE)	1
US Army Aviation Center (ATZQ-D-T, ATZQ-TSM-A, ATZQ-TSM-S, ATZQ-TSM-U)	4
US Army Combined Arms Center (ATZLCA-DM)	1
US Army Safety Center (IGAR-TA, IGAR-Library)	2

US Army Research and Technology Laboratories (DAVDL-AS, DAVDL-POM (Library))	2
US Army Research and Technology Laboratories/Applied Technology Laboratory (DAVDL-ATL-D, DAVDL-Library)	2
US Army Research and Technology Laboratories/Aeromechanics Laboratory (DAVDL-AL-D)	1
US Army Research and Technology Laboratories/Proplusion Laboratory (DAVDL-PL-D)	1
Defense Technical Information Center (DDR)	12
US Military Academy (MADN-F)	1
MTMC-TEA (MTT-TRC)	1
ASD/AFXT	1
Project Manager Black Hawk (DRCPM-BH)	5
US Army Material Development and Readiness Command (DRCDE-SA, DRCQA-ST)	4
US Army Aviation Development Test Activity (STEBG-CT)	2
US Army Troop Support and Aviation Material Readiness Command (DRSTS-WC)	2
US Army Infantry Center (ATSH-TSM-BH)	2
US Army Material Systems Analysis Agency (DRXSY-AAM)	2
US Army Operational Test and Evaluation Agency (CSTE-TM-AV)	2
General Electric (Mr. Koon)	2
Sikorsky Aircraft Division (Mr. Richard Connor)	5