(12)

USAAEFA PROJECT NO. 80-17-1

AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST
PART 1
YAH-64 Advanced ATTACK HELICOPTER

FINAL REPORT

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SEPTEMBER 1981

UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
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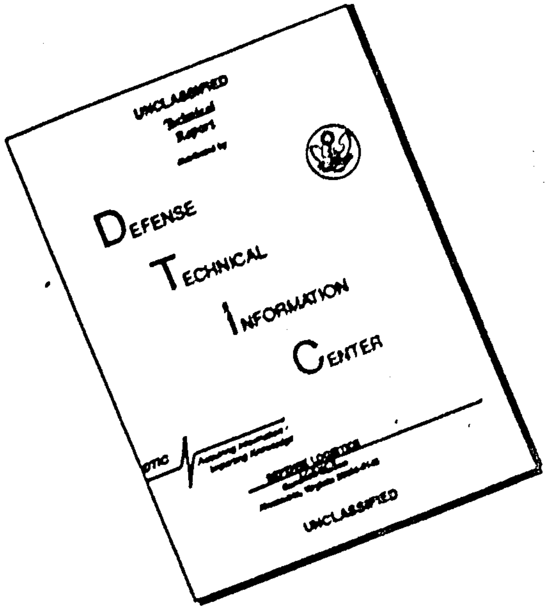

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REPORT DOCUMENTATION PAGE

READ INSTRUCTIONS BEFORE COMPLETING FORM

1. REPORT NUMBER
USAAEFA NO. 80-17-1

2. GOVT ACCESSION NO.
AD A118490

3. RECIPIENT'S CATALOG NUMBER

4. TITLE (and Subtitle)
AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST PART 1 YAH-64 ADVANCED ATTACK HELICOPTER

5. TYPE OF REPORT & PERIOD COVERED
FINAL REPORT
30 May - 17 July 1981

6. PERFORMING ORG. REPORT NUMBER

7. AUTHOR(s)

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8. CONTRACT OR GRANT NUMBER(s)

9. PERFORMING ORGANIZATION NAME AND ADDRESS
US ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AFB, CA 93523

10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS
46-1-P-2117-01-46-EC

11. CONTROLLING OFFICE NAME AND ADDRESS
US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
4300 GOODFELLOW BOULEVARD
ST. LOUIS, MO 63120

12. REPORT DATE
SEPTEMBER 1981

13. NUMBER OF PAGES

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 Test
Performance Qualities
Horizontal Stabilator
Prototype YAH-64 Helicopter
Handling Qualities
Yaw SAS Hardovers

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

The Airworthiness and Flight Characteristics Test Part 1 of the prototype YAH-64 helicopter (S/N 77-23258) was conducted at three test sites; Palomar Airport, Carlsbad, California (elevation 328 ft), Bishop Airport, Bishop, California (elevation 4120 ft), and Coyote Flats, California (elevation 9980 ft). Approximately 32 productive hours were flown during 45 flights between 30 May and 17 July 1981. Several design changes affecting performance and handling qualities were made to the YAH-64 since the last evaluation (EDT 4) the most significant of which were the change in tail rotor rigging, the addition of a third DC electrical bus and the incorporation of a Nap-of-the-Earth/Approach mode of operation for the horizontal stabilator. Level flight performance and vibration levels have been slightly degraded since EDT 4, while handling qualities and

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

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SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered) out-of-ground-effect hover performance remain essentially the same. Numerous anomalies in the operation of the Digital Automatic Stabilization Equipment (DASE) were experienced which may render some of the handling qualities test results suspect. Two deficiencies (both DASE related) were found during this evaluation: Yaw SAS hardovers, and disengagement of the DASE with failure of the No. 2 generator. Nine previously unreported shortcomings (4 DASE related) were found.

ADP DATA TRANSMITTAL RECORD

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1. SECURITY CLASSIFICATION: UNCLASSIFIED

2. SHIPMENT NO: N/A

3. TITLE/FILE IDENTIFICATION
Airworthiness and Flight Characteristics Test, Part 1, YAH-64 Attack Helicopter

4. AS OF DATE			5. DATE OF SHIPMENT		
YEAR	MONTH	DAY	YEAR	MONTH	DAY
83	05	10	83	05	1

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As per your request

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- ☐ PARCEL POST
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8. REQUIREMENT CONTROL SYMBOL (AR 335-15)

9. NO. BOXES

10. NO. ITEMS/REELS

11. SHIPPED FROM:

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ATTN: DAVTE-P
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14. PERSON TO CONTACT/PHONE
Beverly J. Zarvos/(805)277-4024/4027

15. TYPED NAME AND TITLE OF SENDER
Beverly J. Zarvos, Editorial Assistant

16. SIGNATURE OF SENDER
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17. TYPE COMPUTER AND RECORDING DEVICE

18. REMARKS
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SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 80-17-1, Airworthiness and Flight Characteristics Test, Part 1, YAH-64 Advanced Attack Helicopter

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1. The purpose of this letter is to establish the Directorate for Development and Qualification position on the subject report. The objectives of this test were to assess the handling characteristics, performance and vibration characteristics of the YAH-64 helicopter upon completion of full scale engineering development. Due to schedule restraints, the test was divided into three parts, of which this is the first.

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

a. Paragraph 52d. The No. 3 DC electrical buss was incorporated to prevent unacceptable voltage transients in flight critical systems following a failure in either the No. 1 or No. 2 electrical buss. This modification was incorporated in all four prototypes, and was tested successfully in the three other aircraft. It is believed the unsuccessful implementation is unique due to instrumentation, installation and peculiar electrical system for this test aircraft.

b. Paragraph 53a. Yaw SAS hardover failures were caused by failure of a power supply internal to the DASE. The power supply output voltage was monitored for total failure, but a "soft" failure where an incorrect voltage was output would not have been detected. The voltage monitor threshold will be changed to detect off-voltage conditions as well as full failures.

c. Paragraph 53b. Disengagement of the DASE following failure of the No. 2 generator is peculiar to this aircraft. The other prototype aircraft do not exhibit this characteristic with the addition of the No. 3 DC electrical buss.

d. Paragraph 54a. The extraneous illumination of the Master Caution light was caused by the reset switching mechanism. A different design will be incorporated for production.

e. Paragraph 54b. The APU configuration installed in the aircraft did not have all changes defined during qualification testing. The APU start system, used for the APU qualification tests, successfully demonstrated APU starts throughout all required environmental conditions. With the production design, the reported APU start failures should be eliminated.

f. Paragraph 54c. The environmental control system is currently under redesign to provide adequate cooling/heating to the crew and avionics.

g. Paragraph 54d and f. The erroneous activation of the engine out/main rotor RPM and stabilator audio warning tone, with failure of the No. 1 or No. 2 generator is unique to the test aircraft as discussed previously in this letter.

h. Paragraph 54g. The persistent yaw oscillations were attributed to the DASE power supply failure previously mentioned. The phenomenon will be evaluated during A&FC Part 2.

i. Paragraph 54h. The stabilator system contains a feature to switch from manual mode to automatic mode as the aircraft accelerates above 80 knots. There was an anomaly in the programming which resulted in inconsistent switching to automatic mode when the stabilator was at 35 degrees LEU. The monitor program was altered to correct the discrepancy and will be checked in A&FC Part 2.

j. Paragraph 55. The position on the shortcomings identified in this paragraph is reported in the EDT 4, and is unchanged.

FOR THE COMMANDER:

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

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INTRODUCTION

BACKGROUND

1. In June 1973, the United States Army Aviation Systems Command, since renamed the US Army Aviation Research and Development Command (AVRADCOM), awarded a Phase I Advanced Development Contract to Hughes Helicopters (HH). The contract required HH to design, develop, fabricate, and initiate a development/qualification effort on two Advanced Attack Helicopter prototypes and a ground test vehicle as part of a Government Competitive Test. The United States Army Aviation Engineering Flight Activity (USAAEFA) conducted Development Test I using two of these aircraft (ref 1, app A). In December 1976, AVRADCOM awarded a phase II Engineering Development Contract to HH for further development and qualification of the YAH-64 to include subsystems and mission essential equipment. During this program, Engineer Design Tests (EDT) 1 and 2 were conducted by USAAEFA to evaluate developmental progress (refs 2 and 3). In December 1980 USAAEFA conducted EDT 4 to evaluate the performance and flight handling characteristics of a new empennage configuration (ref 4). In December 1980 and January 1981 USAAEFA, in conjunction with US Army Aviation Development Test Activity, conducted EDT 5 to assess the integrated operation of all YAH-64 subsystems (ref 5). AVRADCOM requested (ref 6) USAAEFA to conduct this airworthiness and flight characteristics (A&FC) test which follows incorporation of changes to improve flying qualities, structural integrity, flight performance, and vibration characteristics of the prototype aircraft. Additional changes are planned for the production aircraft. A test plan (ref 7) was submitted in March 1981, and an airworthiness release (ref 8) was issued in May 1981 and revised in June 1981.

TEST OBJECTIVES

2. The objectives of this Part 1 A&FC test were to assess the handling characteristics, hover, vertical climb, and level flight performance, and to compare vibration characteristics with those from previous testing. A follow-on Part 2 A&FC test, which will complete the A&FC phase of testing, is scheduled in December 1981.

DESCRIPTION

3. The YAH-64 is a two-place, tandem seat, twin-engine helicopter with four bladed main and antitorque rotors, conventional wheel landing gear and a movable horizontal stabilator. The helicopter is powered by two General Electric YT700-GE-700R turboshaft engines rated at 1563 shaft horsepower (SHP) (sea level standard day, uninstalled). A 30mm chain gun is mounted on the underside of the fuselage below the front cockpit. The helicopter has wings with two store pylons on each side to carry either HELLFIRE missiles, 2.75-inch folding fin aerial rockets, auxiliary fuel tanks or a combination thereof. An aerodynamic mockup of the Martin-Marietta Target Acquisition Designation System/Pilot Night Vision System was installed. The test helicopter was HH air vehicle number 5 (US Army serial number 77-23258). Major changes to the helicopter since EDT-4 include:

a. A third mode (Nap-of-the-Earth(NOE)/Approach) for the horizontal stabilator.

b. Digital Automatic Stabilization Equipment (DASE) software modifications:

(1) Reduction in pitch Command Augmentation System (CAS) and Stability Augmentation System (SAS) gains

(2) Reduction of pitch CAS authority with airspeed

(3) Sideslip input to DASE provided by Air Data System (ADS)

c. Vibration absorber removed

d. Full left tail rotor rigging at 27 degrees blade angle

e. Addition of a third direct current (DC) electrical bus

f. External configuration changes affecting drag:

(1) Stiffened wings

(2) External stringers for increased tail boom strength

(3) Removal of wing tip fairings

(4) Installation of rotating pulse code modulation (PCM) instrumentation canister and antenna brackets

(5) Instrumented main and tail rotor blades

Further description of the helicopter is contained in the operator's manual and appendix B.

TEST SCOPE

4. Flight testing for A&FC (Part 1) was conducted at three test sites. A total of 45 flights were conducted for a total of 32.4 productive flight hours. Twenty flights were conducted at Palomar Airport, Carlsbad, California (328 ft elevation) from 30 May through 7 June 1981, and from 26 June through 17 July 1981. Twenty flights were conducted at Bishop Airport, Bishop, California (4120 ft elevation) and at Coyote Flats, California (9980 ft elevation) from 10 June through 25 June 1981. Three Army pilots performed the evaluation. However, during low-speed flights conducted at Bishop and Coyote Flats (3 flights), a HH pilot flew as aircraft commander in the copilot gunner seat. HH installed, calibrated and maintained the test instrumentation and performed all aircraft maintenance during the test. Flight restrictions and operating limitations contained in the airworthiness release issued by AVRADCOM and the operator's manual (ref 9, app A) were observed during the evaluation. Where possible, flight test data were compared to the system specification and results obtained during EDT-4. The test conditions are shown in table 1.

Table 1. Test Conditions[1]

Type of Test	Avg Gross Weight (lb)	Avg Longitudinal CG (FS)	Avg Density Altitude (ft)	Trim Calibrated Airspeed (KT)	Stabilator Mode	Wing Stores
HOVER PERFORMANCE	13720	204.5 (Aft)	6060	--	Auto	Clean
	13440	206.2 (Aft)	11580	--	Auto	Clean
LEVEL FLIGHT PERFORMANCE	16100	201.3 (Fwd)	8880	76-141[2]	Auto	16 HELLFIRE
	15240	202.1 (Fwd)	8180	51-149[2]	Auto	8 HELLFIRE
STATIC LONGITUDINAL STABILITY	15560	204.4 (Aft)	6840	59 and 128	Auto	16 HELLFIRE
MANEUVERING STABILITY	15480	204.4 (Aft)	7180	126	Auto	16 HELLFIRE
LOW SPEED[2]	17860	201.3 (Fwd)	820	35 RT	Auto	16 HELLFIRE
	16320	201.9 (Fwd)	5740	45 Fwd-Rwd 45 LT 40 RT	Auto	16 HELLFIRE
	15060	202.3 (Fwd)	6620	45 Fwd-Rwd 45 LT-RT	Auto	8 HELLFIRE
	14840	202.5 (Fwd)	10580	45 Fwd-Rwd 45 LT-RT	Auto	8 HELLFIRE
STABILATOR EVALUATION:						
Control Positions	14960	202.5 (Fwd)	700	45 Fwd-Rwd[2] 45 LT-RT[2]	MAN[3]	8 HELLFIRE
	15180	202.1 (Fwd)	8000	52-120	MAN	8 HELLFIRE
Level, Climb, Descent	15240	202.2 (Fwd)	6440	80 ± 10	NOE[4]	8 HELLFIRE
Static Longitudinal Stability	14800	202.1 (Fwd)	7940	73	NOE	8 HELLFIRE
Acceleration/ Deceleration	14920	202.6 (Fwd)	640	0-125	NOE & MAN[5]	8 HELLFIRE
Malfunctions	15100	202.6 (Fwd)	780	0-115	MAN	8 HELLFIRE
Approach/ Landing	14920	202.6 (Fwd)	780	96-0	Auto & NOE & MAN[5]	8 HELLFIRE
DASE EVALUATION	14900	202.5 (Fwd)	580	0-98	Auto	8 HELLFIRE

NOTES:

[1] Rotor speed 289 RPM (100%), except hover performance
[2] Airspeed presented is true airspeed
[3] Manual - Stabilator angle 25 & 35 degrees leading edge up (LEU)
[4] NOE: Nap-of-the-earth/Approach
[5] Manual - Stabilator angle 35 degrees LEU

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures were used (refs 10 and 11, app A). Test methods are briefly discussed in the Results and Discussion section of this report. A vibration rating scale (VRS) (fig. 1, app D) was used to augment crew comments relative to aircraft vibration levels. A handling qualities rating scale (HQRS) (fig. 2) was used to supplement pilot comments on the handling qualities. Flight test data were obtained from calibrated test instrumentation and were recorded on magnetic tape. Real time telemetry was used to monitor selected parameters throughout the flight test. A detailed listing of the test instrumentation is contained in appendix C. Data analysis methods are described in appendix D.

RESULTS AND DISCUSSION

GENERAL

6. Several design changes affecting performance and handling qualities (para 3) were made to the YAH-64 since the last evaluation (EDT-4). Level flight performance and vibration levels have been slightly degraded since EDT-4, while handling qualities and out-of-ground-effect (OGE) hover performance remain essentially the same. Numerous anomalies in the operation of the DASE were experienced during this evaluation. These DASE related problems may render some handling qualities test results suspect. Two deficiencies (both DASE related) were found during this evaluation: yaw SAS hardovers, and disengagement of the DASE with failure of the No. 2 generator. Nine previously unreported shortcomings (four DASE related) were also found.

PERFORMANCE

General

7. In-ground-effect (IGE) and OGE hover and level flight performance were measured during this program. The drag of the aircraft has increased since EDT-4 (para 3). Vertical climb performance tests were attempted, but inaccuracies in the low airspeed system rendered the data useless. Engine installation losses necessary to calculate installed power available were not determined during this test, therefore, the performance requirements of the system specification (ref 12, app A) could not be evaluated. Performance tests were flown at the conditions listed in table 1.

Hover Performance

8. Hover performance was evaluated at 5- and 100-foot wheel heights using tethered and freeflight techniques. Hover performance data, in nondimensional form, are presented in figures 1 and 2, appendix E. OGE hover performance was essentially unchanged from EDT-4 at low C_T's and slightly better at high C_T's. The YAH-64 requires 2142 SHP to hover OGE and 1831 SHP to hover IGE (5 ft) at a gross weight of 14,525 pounds on a 4000-foot, 95-degree fahrenheit day.

Vertical Climb Performance

9. Vertical climb performance was evaluated using ADS information supplied to a cockpit low airspeed indicator to maintain zero lateral and longitudinal airspeed during the climb. Climbs were flown in surface winds of 3 knots or less. Although the cockpit low airspeed indicator showed less than 3 knots, the aircraft appeared to translate fairly rapidly to the rear during the climbs. This was assumed to be caused by an increase in wind velocity with altitude. Analysis of the data indicated the actual power required during vertical climbs was considerably less than indicated by momentum theory analysis. Further investigation (comparison of low airspeed pace vehicle data with ADS data at the same conditions) indicated that the aircraft could be translating as much as 15 knots rearward when the ADS system was indicating less than 3 knots (figs. 83 and 84, app E). This suggests that during the vertical climbs the aircraft was probably flying rearward and thus using less power than required for vertical climb. Vertical climb performance should be re-evaluated during Part 2 A&FC using an accurate low-airspeed system.

Level Flight Performance

10. Power required for level flight was measured in the 8- and 16-HELLFIRE configurations at constant rotor speed in zero sideslip flight. Nondimensional data for the 8-HELLFIRE configuration are presented in figures 3 through 6, appendix E. Figure 7 presents a comparison of performance data from this test with EDT-4 data. This comparison shows that the drag of the aircraft increased between EDT-4 and this test. The change in drag is equivalent to an increase in flat plate area of approximately 3 square feet (assuming a propulsive efficiency of one). An increase in drag was caused by the changes in external aircraft configuration described in appendix B (table 1). At the conditions of figure 7, the power required for level flight at 146 knots true airspeed (KTAS) has increased from 2120 SHP in EDT-4 to 2215 SHP. Figures 8 through 10 present the 8-HELLFIRE data from this test in dimensional form. Figures 11 and 12 present the limited amount of 16-HELLFIRE data gathered during this test. These figures also include power required information for the 8-HELLFIRE configuration at the same conditions (derived from figures 3 through 6).

HANDLING QUALITIES

General

11. The handling qualities of the YAH-64 helicopter were evaluated at the conditions listed in table 1. All tests were conducted using standard flight test techniques (ref 11, app A) and maneuvers were flown at zero sideslip where possible. Tests were performed with the copilot/gunner cyclic control stick in the retracted position. Test results were compared, where possible, with EDT-4 data. Control position data was influenced by frequent off-center SAS actuator position.

Control Positions in Trimmed Forward Flight

12. The control positions in trimmed forward flight were evaluated in level flight and dives for the 8-HELLFIRE configuration and in level flight only for the 16-HELLFIRE configuration. Tests were conducted at zero sideslip with trim Feel ON and Attitude Hold OFF. Data are presented in figures 13 and 14, appendix E. The longitudinal control position variation with airspeed was conventional in that increasing forward cyclic control was required with increasing airspeed. The shallow stick position gradient below 75 knots calibrated airspeed (KCAS) provided virtually no position or force cue for airspeed control but was not a significant problem since precise airspeed control is not required during NOE and contour flight where this airspeed range would be most often used. The shallow stick position gradient over the airspeed range of 85 to 110 KCAS (less than 0.6-inch of longitudinal cyclic) is an airspeed range where precise airspeed control may be required, particularly during flight in instrument meteorological conditions (IMC). The poor trim characteristics within this airspeed range were annoying to the pilot and frequent activation of trim release switch was required to establish a trim airspeed within ±5 knots indicated airspeed (KIAS) of that desired (HQRS 4). The trim characteristics, though satisfactory for flight under visual meterological conditions, should be further evaluated during IMC flight.

Static Longitudinal Stability

13. The static longitudinal stability characteristics were evaluated at zero sideslip in the 16-HELLFIRE configuration with Trim Feel ON and attitude Hold OFF. Data from these tests are presented in figures 15 and 16, appendix E. The longitudinal cyclic control position gradient indicated approximately neutral stability at a trim airspeed of 59 KCAS and was essentially the same as that reported in EDT-4. The static longitudinal stability at a trim airspeed of 127 KCAS, though still weak, was slightly improved from that reported during EDT 4 for the 8-HELLFIRE configuration. Neutral or weakly positive static longitudinal stability, although satisfactory and desirable for the low-speed attack mission, may contribute to increased pilot workload during IMC flight where more precise airspeed control is required.

Maneuvering Stability

14. The maneuvering stability characteristics were evaluated in the 16-HELLFIRE configuration using collective fixed steady left and right turns at a trim airspeed of 126 KCAS. Data are presented in figures 17 and 18, appendix E. The stick-fixed stability (control position versus load factor) was weakly positive up to load factors of 1.5 (fig. 17). Data points could not be stabilized above 1.5 g due to increased pilot work load caused by saturation of the pitch SAS actuator and the resultant loss of pitch rate damping. The time history presented in figure 18 shows the typical results obtained during this test. Saturation of the pitch SAS actuator occurred most frequently between load factors of 1.5 and 1.7 and steady state pitch rates of 6 to 9 degrees per second. When the SAS actuator reached its authority limit, the aircraft exhibited a pitch up or "dig in" tendency which significantly increased pilot work load for airspeed and load factor control (HQRS 6-7). The results of this test were essentially the same as those reported during EDT 4 for the 8-HELLFIRE configuration. The loss of pitch rate damping at load factors greater than 1.5 due to saturation of the pitch SAS actuator is a previously reported shortcoming which still exists.

Dynamic Stability

15. Tests were conducted in conjunction with level flight performance in an effort to isolate the cause of a persistent yaw oscillation of ±2 to 4 degrees. Time histories were recorded with the DASE ON and OFF and with the ADS active and disabled (circuit breaker OUT). Test data are presented in figures 19 through 21, appendix E. Comparison of the three time history plots indicates that the persistent yaw oscillation is driven by the SAS actuator in response to an erroneous sideslip signal provided by the ADS. The yaw oscillations were noticeable at all airspeeds tested but were particularly annoying below 90 KIAS. The persistent yaw oscillation of ±2 to 4 degrees in level flight is a shortcoming.

Low-Speed Flight Characteristics

16. Low-speed flight characteristics of the YAH-64 were evaluated using a calibrated ground pace vehicle as a speed reference. The tests were flow IGE at approximately 20 feet wheel height with surface winds less than 5 knots. Data are presented in figures 22 through 29, appendix E.

Forward and Rearward Flight:

17. The results of forward and rearward flight tests are shown in figures 22 through 24. Figure 26 shows a discontinuity in lateral control position at a hover caused by an unexplained shift in lateral SAS actuator position (para 35). These shifts did not occur abruptly, and were not noticed by the pilot during the tests. Figure 24 shows some minor discontinuities in lateral control position in rearward flight at airspeeds greater than 25 KTAS. These minor discontinuities were caused by crosswind effects. Generally, the pilot was able to maintain precise aircraft control with only minimal compensation (HQRS 3). The pilot work load increased in rearward flight at both high altitude (fig. 23) and heavy gross weight (fig. 24) where the aircraft was operated near the power available limit (HQRS 4). Aircraft control margins were satisfactory throughout and tail rotor torque requirements were well below the maximum continuous limit. The forward and rearward flight characteristics of the YAH-64 are satisfactory.

Sideward Flight:

18. The results of sideward flight tests are shown in figures 25 through 28. Generally, precise aircraft control could be maintained with pilot work load higher in left sideward flight. Pilot work load increased at higher density altitudes and heavier aircraft gross weights (HQRS 4). Aircraft control margins were satisfactory and tail rotor torque requirements were well below the maximum continuous limit. Figure 26 shows some unexplained discontinuities in lateral control position, similar to those experienced in forward and rearward flight (para 17). The sideward flight characteristics of the YAH-64 are satisfactory.

Low-Speed Maneuvering:

19. Directional control step inputs were used to evaluate low-speed maneuverability in left and right sideward flight at 35 KTAS. A representative time history is shown in figure 29. The control inputs were made in the direction opposite to the direction of flight. These tests were conducted with a tail rotor blade angle limit of 27 degrees. In both cases a 15 degree per second yaw rate was generated in 1.5 seconds, which meets the system specification requirement (ref 12, app A). In right sideward flight the maximum yaw rate obtainable was 15 degrees per second which occurred at 1.5 seconds using full left directional control. The test was conducted with the yaw channel of the DASE both on and off with comparable results. Adequate directional control exists with the tail rotor rigged to 27 degrees with full left pedal.

Stabilator Evaluation

20. Tests were conducted to evaluate the characteristics of the stabilator NOE/Approach mode. Additional data were obtained with the stabilator in the automatic mode and in the manual mode at fixed 25 degrees and 35 degrees LEU angles (the maximum allowed stabilator angle). Flights were conducted at Bishop and Palomar airports and were flown at a forward center of gravity (cg) in the 8-HELLFIRE configuration with DASE on. Data are presented in figures A and B and figures 30 through 39, appendix E. A stabilator system description is provided in appendix B.

FIGURE A
SUMMARY OF CONTROL POSITIONS
IN TRIMMED FORWARD AND REARWARD FLIGHT

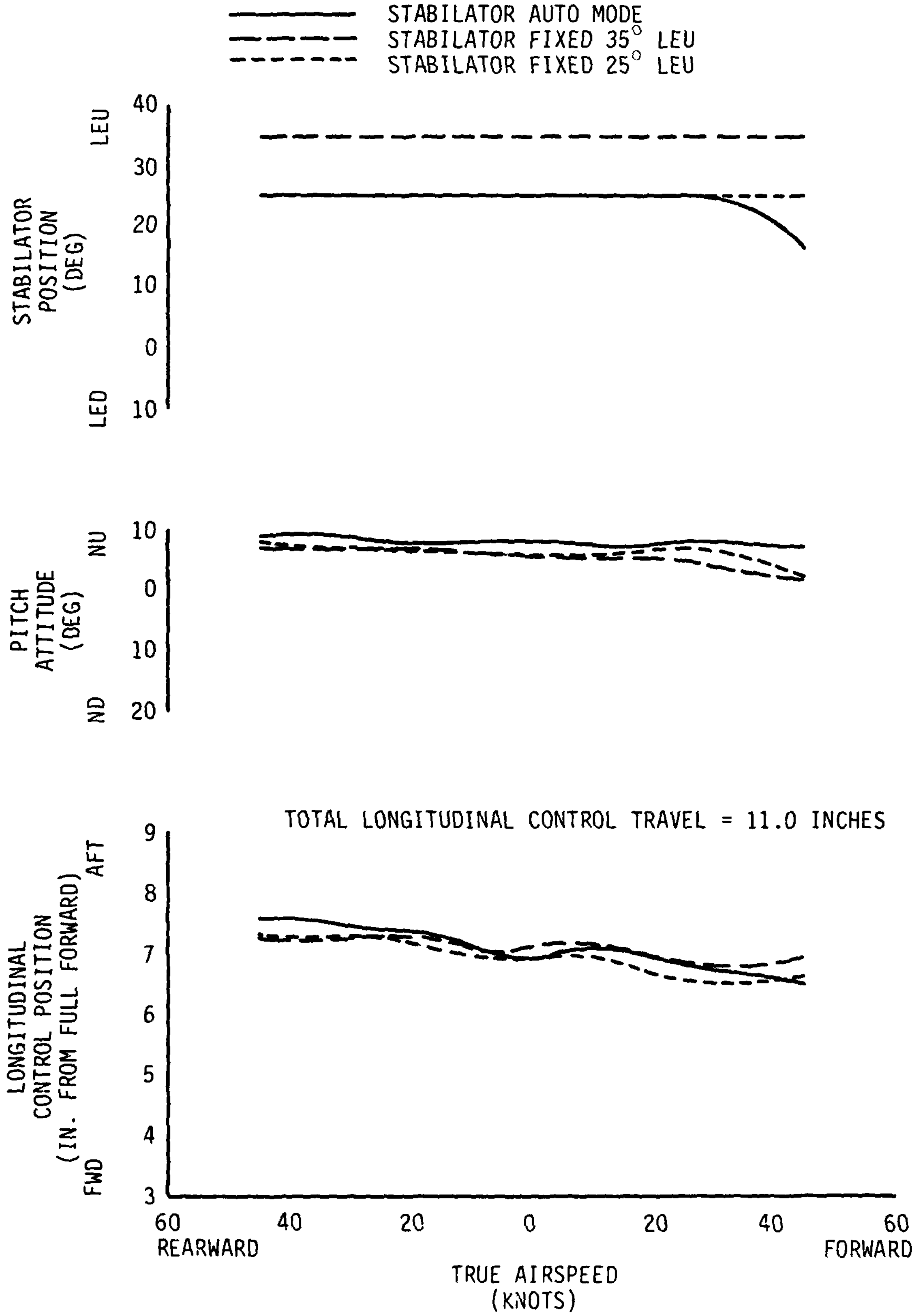
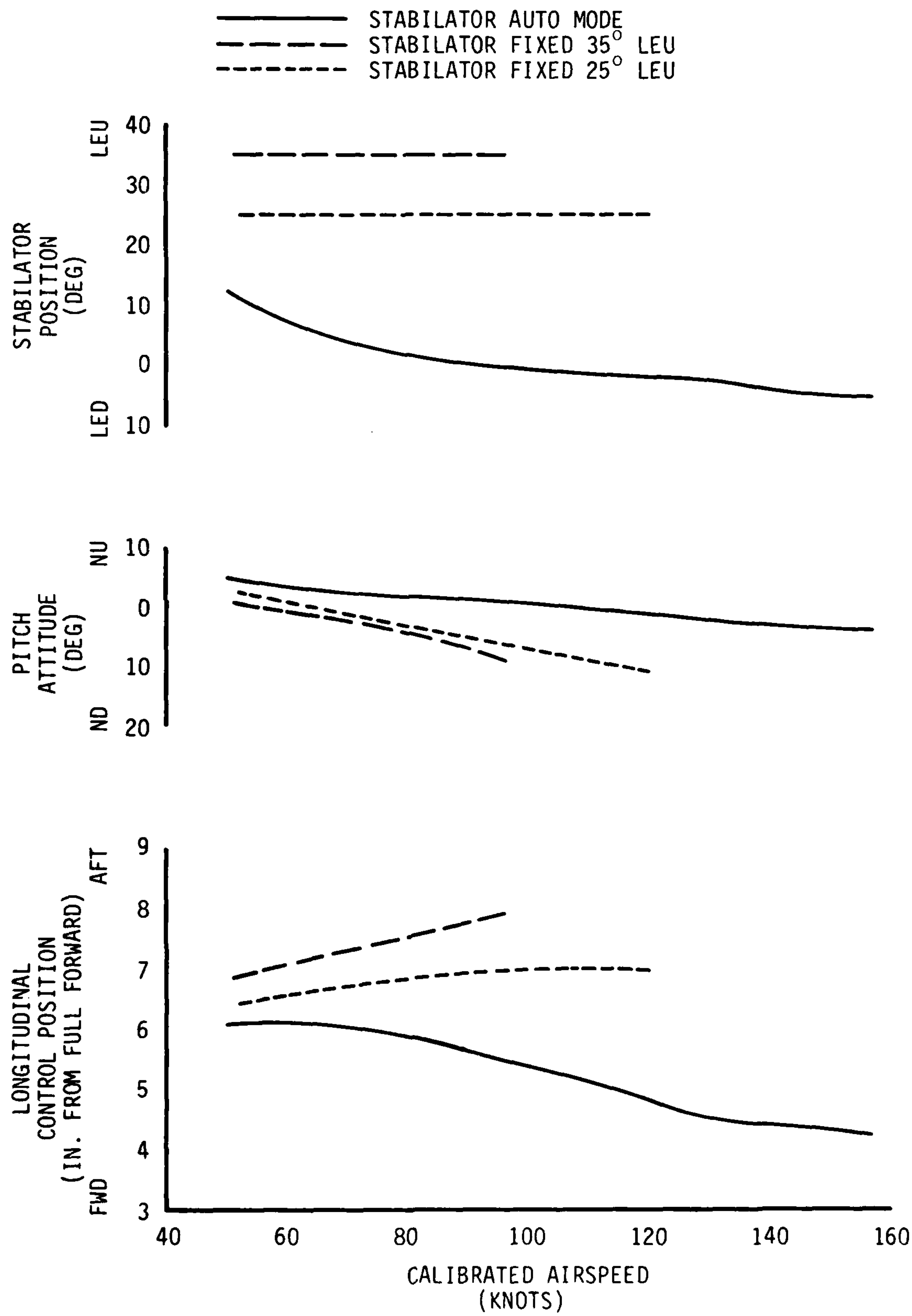


FIGURE B
SUMMARY OF CONTROL POSITIONS
IN TRIMMED FORWARD FLIGHT



Control Positions in Trimmed Forward and Rearward Flight:

21. Control positions in trimmed forward and rearward flight with stabilator angles fixed at 25 and 35 degrees LEU were evaluated from zero to 45 KTAS in 5-knot increments. A ground pace vehicle was used as a speed reference and surface winds were less than 5 knots. The test aircraft wheel height was approximately 20 feet. Data are presented in figures 30 and 31. In general, as stabilator LEU angle increased, nose down pitch attitude and aft cyclic control position required increased. Other control positions were relatively unaffected by stabilator angle. Aircraft control margins were satisfactory. The longitudinal control position variation with airspeed is nonlinear but satisfactory. The small variations from trim in longitudinal, lateral, and directional control movements at all airspeeds indicates the low pilot work load (HQRS 3) in low-speed forward and rearward flight. The discontinuity in lateral control position at a hover was due to an unexplained shift in lateral SAS actuator position (para 35).

22. Control positions in trimmed level flight were evaluated from 50 to 120 KCAS with the stabilator angle fixed at 25 degrees LEU and from 50 to 96 KCAS with the stabilator angle fixed at 35 degrees LEU. Data are presented in figures A and B, and figure 32, appendix E. Results were similar to low-speed flight results (para 17), in that as stabilator LEU angles increased, nose down pitch attitude and aft cyclic requirements increased. Aircraft control margins were satisfactory . Increasing aft cyclic was required to fly increasing airspeeds (one-inch increase between 50 and 95 KCAS at 35 degrees LEU, and 0.6-inch increase at 25 degrees LEU) however, this was not objectionable.

Level Flight, Climbs, and Descents:

23. Level flight, intermediate rated power (IRP) climbs, and minimum power descents were conducted with the stabilator in the NOE/Approach mode. Minimum power descents were initiated from both level flight and IRP climbs by reducing collective to a minimum setting which would maintain 100 percent rotor speed. IRP climbs were initiated from level flight and minimum power descents. During these maneuvers airspeed was initially stabilized slightly below the stabilator switching velocity (80 KIAS) then varied ±10 knots. Increasing airspeed through the switching velocity resulted in a nose up pitch attitude as the stabilator began automatic programming. Reducing airspeed through the switching velocity resulted in a nose down pitch attitude as the stabilator began programming to the NOE/Approach position (25 degrees LEU). The rate and magnitude of these pitch attitude changes were essentially the same during level flight, climbs, and descents. The undesired pitch attitude changes with slight airspeed variations will increase pilot work load during continuous flight at airspeeds close to the stabilator switching velocity. The following note should be placed in the YAH-64 operator's manual:

NOTE

> Continuous flight at airspeeds close to the stabilator switching velocity (80 KIAS ±10 knots) with the stabilator in the NOE/Approach mode will result in aircraft pitch attitude variations caused by intermittent stabilator programming.

Static Longitudinal Stability:

24. [illegible] static longitudinal stability characteristics were evaluated at a trim airspeed [illegible] KCAS with the stabilator in the NOE/Approach mode. The data is provided in [illegible] 33. The longitudinal cyclic control position gradient indicates negative [illegible] with increasing airspeed) at airspeeds below the stabilator [illegible] and positive stability at airspeeds greater than the stabilator [illegible]. The static longitudinal stability is satisfactory for the attack [illegible] although it failed to meet the requirements of paragraph 10 [illegible] 1.1 of [illegible] appendix A, in that there was negative static stability at airspeeds [illegible]

Level Flight Accelerations/Decelerations:

25. To verify proper stabilator programming, level flight accelerations and decelerations through the stabilator switching velocity were conducted in the stabilator's three modes of operation. IRP accelerations from a 20-foot hover were performed with the stabilator in manual mode (35 deg LEU), NOE/Approach mode (25 deg LEU), and in the automatic mode. Representative time histories are presented in figures 34 through 36, appendix E. The stabilator programmed properly in the automatic and NOE/Approach modes. However, stabilator programming in the manual mode was inconsistent. The stabilator, when set to manual or NOE/Approach mode, is designed to switch to automatic programming following acceleration through the stabilator switching velocity (80 KIAS). A monitor circuit is provided to disengage the automatic programming if stabilator angle exceeds 25 degrees LEU when airspeed is greater than 80 KIAS one second after accelerating through the switching velocity. This automatic stabilator shutdown was encountered during 2 of 4 IRP accelerations with the stabilator set to 35 degrees LEU. There were no control margin problems or undesirable aircraft characteristics associated with either complete automatic stabilator programming or stabilator shutdown during IRP accelerations. The stabilator fail audio warning tone provided adequate warning of stabilator shutdown; however, the shutdown served no useful purpose and increased pilot work load by necessitating manual stabilator programming. Failure of the stabilator to consistently achieve complete automatic programming from a full LEU manual setting (35 deg LEU) when accelerating through the stabilator switching velocity is a shortcoming. The stabilator programming should be modified to provide consistent automatic programming from all stabilator settings during acceleration through the stabilator switching velocity.

Stabilator Malfunction During Acceleration:

26. Stabilator malfunction during acceleration was simulated by conducting level flight IRP accelerations from a 20-foot hover at fixed stabilator angles of 25 and 35 degrees LEU. Once set, stabilator angles were fixed by engaging the stabilator "kill" switch located on the pilot's collective lever. Control margins remaining and pilot corrective action required following the simulated failure were evaluated. Representative time histories are presented in figures 37 and 38. Pilot corrective action following the simulated failure required deceleration to avoid exceeding a stabilator airspeed limit. This was accomplished by applying aft cyclic and slightly lowering the collective. Corrective action was initiated two seconds after activation

of the stabilator aural warning. The aural warning which occurred at the stabilator switching velocity (80 KIAS) provided excellent pilot cue of stabilator failure. The minimum control margins remaining during the recovery were 20 percent longitudinal cyclic and 12 percent directional pedal. No undersirable aircraft characteristics or control margins were associated with stabilator failures during IRP accelerations.

Contour Approach/Landing:

27. Contour approaches terminating with a quickstop maneuver and IGE hover were conducted from an initial airspeed of 90 KIAS (stabilator switching velocity + 10 knots) with the stabilator in the NOE/Approach mode. A typical time history is presented in figure 39. The stabilator programmed from 5 degrees leading edge down to 25 degrees LEU in approximately 6 seconds as the aircraft decelerated through the stabilator switching velocity. Maximum nose up pitch attitude during the quickstop was 25 degrees. This pitch attitude restricted pilots forward field of view; however, the pilot could maintain desired heading using peripheral vision. The minimum control margin remaining during the maneuver was approximately 10 percent directional control. Overall pilot workload during transition from contour flight to hover was minimal (HQRS 3). Contour approaches to an IGE hover with the stabilator in the NOE/Approach mode are satisfactory.

Aircraft Systems Failures

Stabilator Failures:

28. Throughout the tests, low power descents at various airspeeds were made. Numerous failures of the stabilator automatic mode of operation were experienced while in a low power descent at airspeeds of 60 to 80 KCAS. The stabilator automatic mode of operation could be regained by depressing the reset button; however, a subsequent automatic mode failure often occurred within a few seconds. The ADS processor was changed (Equipment Performance Report (EPR) No. 80-17-1-5) in an attempt to correct this problem; however, the failures still occurred. The failure of the stabilator automatic mode of operation while in a low power descent is a previously reported shortcoming which still remains.

Auxiliary Power Unit Failures:

29. The auxiliary power unit (APU) was used to start the aircraft engines during this test program. The APU incorporates an automatic start sequence, which is initiated by placing the APU switch from OFF to START and then allowing it to return to RUN. However, the APU failed to start on numerous occasions requiring the utility hydraulic accumulator to be charged in order to attempt a second start. The start procedure was modified to place the APU switch in the RUN position momentarily, allowing the APU electric fuel pump to supply fuel to the APU prior to the start attempt. This procedure reduced the frequency of the starting problems; however, starting failures were not totally eliminated. All the start failures were experienced at the Palomar test site (328 ft elevation). None were experienced at the Bishop test site (4120 ft elevation). The APU fuel control was replaced prior to the

last three test flights (EPR No. 80-17-1-3, app G) and no start failures were experienced at the Palomar test site on six subsequent start attempts. The failure of the APU to start is a shortcoming.

Stability Augmentation System Failures:

30. Throughout the test program multiple SAS hardover failures in the yaw axis were experienced. The hardovers occurred during taxi, hover, takeoff, low power descent, and approach. Figures 40 through 42, appendix E are representative time histories of yaw hardovers. The hardovers were multiple cycle and full SAS actuator travel (20 percent of main flight control servoactuator). These hardovers produced yaw rates up to 23 degrees per second and sideslip angles of 40 degrees at 71 KTAS (fig. 40), which exceeds the sideslip envelope by 7 degrees. Additionally, the SAS failure monitor circuitry did not prevent the hardovers. The ADS processor was changed in an attempt to correct this problem; however, the hardovers were still experienced. Subsequently, the sideslip signal from the ADS to the DASE was removed and no further yaw SAS hardovers were experienced. The yaw rate of 23 degrees per second with a yaw SAS hardover failure failed to meet the requirements of paragraph 10.3.2.7.1 of reference 12, appendix A by 13 degrees per second. The yaw SAS hardover failures (20 percent of main flight control servoactuator travel) are a deficiency which should be corrected prior to Part 2 of the A&FC. The SAS hardover monitor circuitry should be modified to provide complete SAS hardover protection.

Engine Compressor Stalls:

31. During IGE hover while performing low-speed flight tests at 9580 feet pressure altitude, 4.0°C outside air temperature, and 14,920 pounds gross weight, engine compressor stalls were encountered. The aircraft was being operated near engine power limit. The first engine compressor stall occurred on the number two engine while in an IGE hover following a small collective and aft cyclic input. A popping noise was heard and a transient maximum turbine gas temperature (TGT) of approximately 900°C was noted. The collective was lowered and the aircraft landed. Multiple compressor stalls on both engines occurred following a left directional control input of less than one-half inch to arrest a mild right yaw rate. The collective was lowered and the helicopter landed. Popping noises were heard during the descent and engine transient TGT's of 900°C were noted on both engines. Following the flight, the number two engine was changed (EPR No. 80-17-1-2, app G) and no further compressor stalls were encountered. However, conditions which produced the initial stalls were not duplicated. The engines in the helicopter were pre-production engines. The production engines should be evaluated at high density altitude and heavy gross weight to insure adequate engine stall margins.

Generator Failures:

32. Generator failures were simulated in flight by turning off either the No. 1 or No. 2 generator switch. A third DC electric bus had been installed in the aircraft to prevent system failures and erroneous warning activation with the failure of a single generator. Failure of the No. 2 generator caused disengagement of the DASE. DASE disengagement results in loss of SAS, CAS, hover augmentation system (HAS), turn coordination and attitude hold. Since the actuators frequently operate within a range that is near or on one limit of their authority (para 38), a No. 2 generator failure could result in an abrupt, uncommanded control input which would occur as

the actuator centered following DASE disengagement. This problem could occur in all three axes simultaneously and would be of the same magnitude as SAS actuator displacement from trim. Abrupt, uncommanded control inputs, which could be equivalent to 10 percent control authority, will increase the probability of resulting loss of aircraft control during night NOE operations. DASE disengagement with failure of one generator failed to meet the requirement of paragraph 10.3.2.7.8 of reference 12, appendix A. Disengagement of the DASE following failure of the No. 2 generator is a deficiency that should be corrected prior to Part 2 of the A&FC.

33. Simulated failure of the No. 1 generator produced erroneous activation of the stabilator automatic mode failure audio warning tone. The stabilator continued to program in accordance with the automatic schedule. The erroneous activation of the stabilator audio warning tone with failure of the No. 1 generator is a shortcoming.

34. In addition, failure of the No. 1 or No. 2 generator caused erroneous activation of the engine out/main rotor RPM audio warning tone. This could cause the pilot to react to a false engine out situation and compromise mission accomplishment or completion. The erroneous activation of the engine out/main rotor RPM audio warning tone with failure of the No. 1 or No. 2 generator is a shortcoming.

Digital Automatic Stabilization Equipment (DASE) Evaluation

35. During the course of this test program, several anomalies in the operation of the DASE were noted. These consisted of SAS hardovers in yaw (para 30), persistent yaw oscillations in level flight (para 15), loss of SAS rate damping (para 38), and unexplained trim shifts in low-speed flight (paras 17, 18, 21). Other areas of DASE malfunctions occurred in the operation of the HAS (para 36) and the attitude hold system (para 37). The DASE computer was removed following the completion of flight testing on 1 July 1981, and returned to the vendor. A malfuction (undervoltage) in the computer internal power supply was found and corrected. This undervoltage condition reportedly caused a deviation in all the DASE inputs, gains, and washouts. Additional tests were flown to re-evaluate the problem areas related to the DASE after this malfunction was corrected. A modified trim feel system was also evaluated during this DASE reevaluation. The trim feel modification consisted of removing the discrete OFF capability of the force feel system. Due to a related problem with the SAS monitor circuits and the ADS (para 15), the sideslip signal from the ADS to the DASE was removed. This prevented a re-evaluation of yaw SAS hardovers and persistant yaw oscillations in level flight.

36. The HAS was evaluated in a stabilized hover, during bob up, and while repositioning with and without retrimming. The evaluation was conducted to determine if the DASE maintenance action had corrected the abrupt aircraft response when attempting to reposition with HAS ON. Tests were performed in winds of approximately 10 knots. A time history of a gradual acceleration with HAS ON is provided in figure 43, appendix E. The HAS initially resisted the forward longitudinal control input with opposing SAS actuator movement up to full travel. As airspeed reached 35 KIAS, the HAS automatically disengaged and the SAS actuator returned to the centered position. Pitch attitude excursions throughout the maneuvers were less than ±2 degrees. The abrupt aircraft response previously observed appeared to have been corrected. Further evaluation of the HAS, to include operation in gusty wind conditions, should be conducted during operational testing.

37. Abrupt aircraft response following retrimming with attitude hold ON was frequently experienced prior to the DASE maintenance corrective action. Following the maintenance action, the attitude hold feature was re-evaluated. Tests were performed in level flight by trimming at 110 KIAS, decelerating to 100 KIAS and retrimming. The test was repeated trimming at 110 KIAS, accelerating to 120 KIAS and retrimming. A time history of retrimming at 100 KIAS is shown in figure 44. The attitude hold initially resisted the aft longitudinal control input in deceleration with opposing movement of the pitch SAS actuator. While retrimming, the SAS actuator centered, which resulted in the requirement for forward longitudinal control to maintain the desired airspeed. Several attempts were required to achieve a precise trim condition; however, once the desired trim airspeed was attained, the trim condition was maintained within ±3 KIAS in smooth air. No abrupt aircraft response was noted during this test. It appeared the abrupt aircraft response had been corrected with the DASE maintenance action. Further evaluation of the attitude hold feature should be conducted in conjunction with instrument flight capability testing.

38. Full SAS authority was not always available during this evaluation because the SAS actuators were displaced from their centered positions for long periods of time. Time to washout rate damping SAS inputs was greater than 10 seconds for all three axes. Since gust upsets are very short-term phenomena (*i.e.*, less than one second duration) these washout times are excessive. Any rate that persists for more than two seconds can reasonably be assumed to be intentionally caused by the pilot and should not be damped by the SAS (*i.e.*, the rate damping should be washed out at least this rapidly). In turns, the excessive SAS washout time allows the pitch SAS actuator to reach the limit of its travel while trying to oppose the pitch rate which is necessary to maintain the turn. When this happens (at about 1.5 g) all rate damping is lost and pilot work load increases (para 13). In the absence of CAS, aircraft rate response to step control inputs would be washed out to unacceptably low levels by the SAS. On the YAH-64 the solution was to increase the CAS washout time so that the CAS opposes the SAS following a control input. Figure 46 shows the roll CAS washout. These data were taken on the ground following a lateral cyclic displacement from trim. These long CAS washout times have an undesirable side effect: Whenever the controls are displaced from the last trimmed position and there are no aircraft angular rates present, the SAS actuators are displaced from their centered positions for long periods of time. In fact, there is no CAS washout in flight. Figure 45 shows the roll SAS actuator displacement during 30 KTAS rearward flight with the last trimmed control position being that required for hover. Note that there is no tendency for the roll SAS actuator to recenter. This effectively reduces the rate damping authority of the SAS actuators (sometimes to zero). This effect occurs when lifting off to a hover, when changing airspeeds, during any transient maneuver, or at any stabilized flight condition if the control forces are not trimmed to zero. This degradation of rate damping is particularly detrimental during NOE flight where airspeed changes are made very frequently. It was suggested during this evaluation that the pilot depress the momentary trim release button prior to moving the controls from one trimmed flight condition to another and releasing the button when the new stabilized condition is achieved. This would ensure that the CAS would not cause the problem just discussed. However, depressing the button effectively eliminates the CAS function of the DASE. If the CAS is disabled when the pilot moves the controls, then it cannot augment the pilot control inputs, which it is designed to do. The problems caused by excessive SAS and CAS washout times were apparent

throughout the test program, although the severity was reduced following the DASE maintenance action. The reduction of rate damping authority caused by excessive SAS and CAS washout times is a shortcoming. The DASE software should be modified to provide acceptable gust response, control response, and control sensitivity independent of the force feel system reference.

39. A qualitative evaluation of a modified trim feel system (discrete off function eliminated) was conducted during the DASE re-evaluation. The modification required the pilot to either oppose the control force gradient or retrim when displacing the cyclic from a trimmed position. The location of the momentary trim release switch on the cyclic handgrip and the mechanical functioning of the switch made it difficult to operate. Additionally, activation of the switch effectively eliminated the CAS function of the DASE (para 38). Absence of a discrete off capability on the trim feel system was undesirable since many pilots, based on previous experience in other aircraft, prefer to fly without a force gradient. The discrete off function of the trim feel system should not be eliminated.

VIBRATION CHARACTERISTICS

40. The vibration characteristics of the YAH-64 were evaluated with the vertical vibration absorber removed. Qualitative evaluations were made at the pilot's and copilot's stations. Quantitative vibration data were gathered at the conditions in table 2, and are shown in figures 47 through 80, appendix E.

41. Vibration characteristics were evaluated in forward and rearward low-speed flight with the stabilator in the manual mode at 25 and 35 LEU and in the automatic mode of operation (figs. 47 through 59). The 4/rev (19.3Hz) vertical vibration of 0.4 to 0.55 g at the pilot's seat at airspeeds greater than 25 KTAS rearward flight are objectionable (VRS 6). The use of manual stabiliator full LEU (35°) did not change the vibration characteristics as compared to 25 degrees LEU. Although the vibrations at the copilot's seat were lower than at the pilot's seat for the same flight regime (0.35 to 0.4 g) they were still objectionable (VRS 5).

42. The vibration characteristics in sideward low-speed flight were evaluated with the stabilator in the automatic mode (figs. 63 through 69). The 4/rev (19.3 Hz) lateral vibrations in right sideward flight of 0.3 to 0.36 g at airspeeds above 20 KTAS were objectionable to the pilot (VRS 5). The vibrations at the copilot station were again lower than at the pilot's station and were not objectionable.

43. The vibration characteristics in level flight were evaluated with the stabilator in the automatic mode (figs. 70 through 80). The 4/rev (19.3Hz) lateral vibration greater than 0.2 g at airspeeds below 54 KCAS were objectionable to the pilot (VRS 4). The 4/rev lateral and vertical vibration greater than 0.2 g at airspeeds greater than 128 KCAS were objectionable to the pilot (VRS 4) (fig. 72). The vibrations at the copilot seat were lower than at the pilot seat and were not objectionable.

44. The vibration characteristics during the termination of the approach were qualitatively evaluated. These vibrations were similar to those encountered during rearward flight at airspeeds greater than 25 KTAS and were objectionable to both the pilot (VRS 6) and copilot (VRS 5).

45. The 4/rev vibration characteristics of the YAH-64 were generally degraded from EDT-4. The vibration characteristics of the other harmonics remained essentially unchanged. The vibration characteristics are more severe at the pilot's seat than at the copilot's seat and remain objectionable during several flight regimes. The 4/rev (19.3Hz) vibration characteristics remain a shortcoming.

COCKPIT EVALUATION

46. The environmental control unit (ENCU) was evaluated on all flights. Ambient temperature conditions ranged up to 95°F on clear, sunny days. The ENCU was inadequate in terms of both air flow and air temperature and failed to keep either the pilot or copilot crew station at an acceptable level of comfort. Failure of the ENCU to provide adequate crew station cooling is a shortcoming and should be corrected prior to production. The ENCU should be further evaluated during climatic testing.

RELIABILITY AND MAINTAINABILITY

47. The reliability and maintainability features of the YAH-64 aircraft were evaluated throughout the test. Twenty-two EPR's were prepared and submitted during this test and are listed in appendix G. This section is intended to highlight those undesirable reliability and maintainability features encountered.

48. The master caution light illuminated on at least three occasions without a corresponding caution, warning, or advisory panel segment light. The master caution light is designed to be a latched light while the segment lights are not. On at least two occasions the pilot observed the caution, warning, and advisory panel when the master caution light illuminated. None of the segment lights flickered indicating a momentary fault condition and the segment lights were then subsequently tested to insure all lights would illuminate. These spurious illuminations of the master caution light occurred while taxiing into the parking area and prior to the connection of external electrical power. Illumination of the master caution light without the illumination of a caution, warning or advisory panel segment light is a shortcoming.

49. Throughout the test, the Marconi engine and rotor speed indicator consistently indicated 102 percent (in the caution range) main rotor speed (N_R) and No. 1 and No. 2 engine power turbine speed (N_P). The calibrated flight test instrumentation indicated 100 percent N_R and N_P at the same conditions. The illumination of a caution range is distracting to the pilot since it indicates an overspeed condition. The erroneous indication of 102 percent N_R and N_P is a shortcoming which was previously reported.

MISCELLANEOUS

50. Testing was accomplished to further evaluate deficiencies and shortcomings identified during earlier tests. As a result of this evaluation, two previously reported deficiencies have been corrected and 13 shortcomings have either been corrected or determined to be satisfactory due to modifications of procedures or limitations. The following previously reported shortcomings (in order of importance) remain valid shortcomings. A discussion of these shortcomings can be found in reference 4,

appendix A. The Program Manager and HH have identified corrective action for the shortcomings with asterisks (ref 15, app A).

a. The poor design of the pilot's fuel control panel.*

b. The absence of SAS pitch rate damping at load factors greater than 1.6 due to saturation of the pitch SAS actuator.*

c. The restriction to the pilot's field of view caused by window edge distortion, the overhead circuit breaker panel, canopy reflections, CPG helmet, and the pilot night vision system turret.

d. The excessive longitudinal breakout force (specification noncompliance).

e. The inadvertent directional control inputs during brake application.

f. Dissengagement of the stabilator automatic mode of operation while in a low power descent.*

g. The poor design of the collective pitch control friction mechanism.*

h. The lack of a reliable indication of parking brake status.*

i. The lack of an acceptable method for sampling the primary and utility hydraulic fluid.

j. The high inherent friction of the engine power control levers.

k. The annoying tone present in the intercom system.*

l. The failure of the Marconi instruments to display the full green range during normal operation.

m. The excessive accumulation of oil on the main transmission deck and in the upper fairing maintenance access area.*

n. The poor location of the pilot engine control quadrant.**

o. The washout of the rocket panel display, Marconi instrument indications and caution, warning, and advisory panel segment lights in direct sunlight.

p. The poor location of the tail wheel unlock light.

q. The constant illumination of the lower green segment light on the Marconi vertical scale.

r. The illumination of the APU ON advisory light prior to the APU stabilizing at 100 percent RPM.*

s. The poor anthropometric design of the pilot's cyclic grip.*

t. The difficulty in accurately determining the engine oil levels without opening the engine cowlings.*

u. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls.

v. The illumination of the master caution light with green advisory segment lights.*

w. The failure of the fire bottle discharge lights to illuminate with the activation of the PRESS TO TEST switch.*

AIRSPEED CALIBRATION

51. The left and right pitot-static airspeed systems were calibrated in level flight and dives using both T-28 and AH-1S pace aircraft for the calibrated airspeed reference. Figures 81 and 82, appendix E present data from this calibration effort. The main-rotor-mast-mounted omnidirectional airspeed system was calibrated in low-speed forward, rearward, left sideward and right sideward flight using a pace automobile as a speed reference (figs. 83 and 84). A large dead band exists in the longitudinal measurement of the low airspeed system. At actual airspeeds from -15 knots rearward to 3 knots forward, the longitudinal airspeed output was indicating from 3 knots forward to 3 knots rearward. This affected the gathering of vertical climb performance data during this test. It is not known if this error is significant from an operational standpoint.

CONCLUSIONS

GENERAL

52. Based on the A&FC Part 1 flight test of the YAH-64 helicopter, the following conclusions were reached:

a. Adequate directional control exists with the tail rotor rigged to 27 degrees with full left pedal (para 19).

b. 4/rev vibration characteristics have degraded since EDT-4 (para 45).

c. Manual stabilator setting of 35 degrees LEU did not change the vibration in rearward flight (para 41).

d. Incorporation of the No. 3 DC electrical bus did not prevent loss of DASE with failure of the No. 2 generator (para 32).

e. Twenty-two equipment performance reports were submitted during this test (app G).

f. Two deficiencies have been identified.

g. Ten previously unreported shortcomings have been identified.

h. Two previously reported deficiencies have been corrected.

i. Thirteen previously reported shortcomings have either been corrected or determined to be satisfactory due to modification of procedures or limitations.

DEFICIENCIES

53. The deficiencies reported herein are not necessarily intended to bar type classification as per AR 310-25 (see app D for definition of deficiency used in this report). The following deficiencies (in order of relative importance) were identified:

a. Yaw SAS hardover failures (20 percent of main flight control servo actuator travel) (para 30).

b. Disengagement of the DASE following failure of the No. 2 generator (para 32).

SHORTCOMINGS

54. The following previously unreported shortcomings (in order of relative importance) were identified (see app D for definition of shortcomings used in this report):

a. Illumination of the master caution light without the illumination of a caution, warning or advisory panel segment light (para 48).

b. Failure of the APU to start (para 29).

c. Failure of the ENCU to provide adequate crew station cooling (para 46).

d. Erroneous activation of the engine out/main rotor RPM audio warning tone with failure of the No. 1 or No. 2 generator (para 34).

e. Reduction of rate damping authority caused by excessive SAS and CAS washout times (para 38).

f. Erroneous activation of the stabilator audio warning tone with failure of the No. 1 generator (para 33).

g. Persistent yaw oscillations of ±2 to 4 degrees in level flight (para 15).

h. Failure of the stabilator to consistently achieve complete automatic programming from a full LEU manual setting (35 deg LEU) when accelerating through the stabilator switching velocity (para 25).

i. The 4/rev (19.3 Hz) vertical vibration characteristics (para 45).

55. The following shortcomings (in order of relative importance) were identified during EDT 4 and earlier tests. They were further evaluated during this test and remain valid shortcomings. The Program Manager and HH have identified corrective action for the shortcomings with asterisks (ref 15, app A).

a. The poor design of the pilot's fuel control panel.*

b. The absence of SAS pitch rate damping at load factors greater than 1.6 due to saturation of the pitch SAS actuator.*

c. The restriction to the pilot's field of view caused by window edge distortion, the overhead circuit breaker panel, canopy reflections, CPG helmet, and the PNVS turret.

d. The excessive longitudinal breakout force (specification noncompliance).

e. The inadvertent directional control inputs during brake application.

f. Failure of the stabilator automatic mode of operation while in a low power descent.*

g. The poor design of the collective pitch control friction mechanism.*

h. The lack of a reliable indication of parking brake status.*

i. The lack of an acceptable method for sampling the primary and utility hydraulic fluid.

j. The high inherent friction of the engine power control levers.

k. The annoying tone present in the intercom system.*

l. The failure of the Marconi instruments to display the full green range during normal operation.

m. The excessive accumulation of oil on the main transmission deck and in the upper fairing maintenance access area.*

n. The erroneous indication of 102 percent N_R and N_P (para 49).

o. The poor location of the pilot engine control quadrant.

p. The washout of the rocket panel display, Marconi instrument indications and caution, warning, and advisory panel segment lights in direct sunlight.

q. The poor location of the tail wheel unlock light.

r. The constant illumination of the lower green segment light on the Marconi vertical scale.

s. The illumination of the APU ON advisory light prior to the APU stabilizing at 100 percent RPM.*

t. The poor anthropometric design of the pilot's cyclic grip.*

u. The difficulty in accurately determining the engine oil levels without opening the engine cowlings.*

v. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls.

w. The illumination of the master caution light with green advisory segment lights.*

x. The failure of the fire bottle discharge lights to illuminate with the activation of the PRESS TO TEST switch.*

SPECIFICATION COMPLIANCE

56. The YAH-64 was found to be in noncompliance with the following paragraphs of the Phase II Advanced Attack Helicopter System Specification AMC-SS-AAH-H10000A during these tests.

a. 10.3.2.7.1 Yaw rate in excess of 10 degrees per second with a yaw SAS hardover failure (para 30).

b. 10.3.2.7.8 Failure of the DASE with the loss of the No. 2 generator (para 32).

c. 10.3.4.1 Negative longitudinal static stability with the stabilator in the NOE/Approach made at airspeeds below the stabilator switching velocity (para 24).

RECOMMENDATIONS

57. The following recommendations are made:

a. Correct the deficiencies prior to A&FC Part 2 testing.

b. Correct the shortcoming in paragraph 54g prior to A&FC Part 2 testing.

c. Correct the remaining shortcomings prior to production.

d. The DASE software should be modified to provide acceptable gust response, control response, and control sensitivity independent of the force feel system reference (para 38).

e. The SAS hardover monitor circuit should be modified to provide complete SAS hardover protection (para 30).

f. The discrete off function of the trim feel system should not be eliminated (para 39).

g. Place the following note in the YAH-64 operator's manual (para 23):

NOTE

> Continuous flight at airspeeds close to the stabilator switching velocity (±10 knots) with the stabilator in the NOE/Approach mode may result in aircraft pitch attitude variations caused by intermittent stabilator programming.

h. Further evaluation of the HAS to include operation in gusty wind conditions, should be conducted during operational testing (para 36).

i. Further evaluation of trim characteristics and the attitude hold feature should be conducted in conjunction with instrument flight capability testing (paras 12 and 37).

j. Production engines should be evaluated at high density altitude and heavy gross weight to insure adequate engine stall margins (para 31).

k. The ENCU should be further evaluated during climatic testing (para 46).

l. Modify the stabilator programming to provide consistent automatic programming from all stabilator settings during acceleration through the stabilator switching velocity (para 25).

m. Vertical climb performance should be reevaluated during Part 2 A&FC using an accurate low airspeed system (para 9).

APPENDIX A. REFERENCES

1. Final Report, USAAEFA Project No. 74-07-2, *Development Test 1, Advanced Attack Helicopter Competitive Evaluation, Hughes YAH-64 Helicopter*, December 1976.

2. Final Report, USAAEFA Project No. 77-36, *Engineer Design Test 1, Hughes YAH-64 Advanced Attack Helicopter*, September 1978.

3. Final Report, USAAEFA Project No. 78-23, *Engineer Design Test 2, Hughes YAH-64, Advanced Attack Helicopter*, June 1979.

4. Final Report, USAAEFA Project No. 80-03, *Engineer Design Test 4, YAH-64 Advanced Attack Helicopter*, January 1980.

5. Letter, USAAEFA, DAVTE-TI, 9 March 1981, subject: Report, Engineer Design Test, Government EDT-5 of the Advanced Attack Helicopter (AAH), USAAEFA Project No. 80-12.

6. Letter, AVRADCOM, DRDAV-DI, 1 April 1981, subject: Airworthiness and Flight Characteristics (A&FC) Test of the YAH-64 Advanced Attack Helicopter, Prototype Qualification Test-Government (PQT-6).

7. Test Plan, USAAEFA Project No. 80-17-1, *Airworthiness and Flight Characteristics Test, Part I, YAH-64 Advanced Attack Helicopter*, March 1981.

8. Letter, AVRADCOM, DRDAV-D, 28 May 1981, revised 18 June 1981, subject: Airworthiness Release for Airworthiness and Flight Characteristics Test Part I of YAH-64 Helicopter, S/N 77-23258.

9. Technical Manual, TM 53-1520-238-10, *Operator's Manual for Army YAH-64 Helicopter*, 1 May 1981.

10. Pamphlet, USAMC, AMCP 706-204, "Engineering Design Handbook Helicopter Performance Testing," August 1974.

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

12. Specification, Hughes Helicopters, AMC-SS-AAH-H100000A, "YAH-64, Phase II Advanced Attack Helicopter Systems," 10 December 1976.

13. System Description, Hughes Helicopters, "YAH-64, Stabilator Control, Phase II," 4 August 1981 (preliminary).

14. System Description, Hughes Helicopters, "YAH-64 Digital Automatic Stabilization Equipment, Phase II," 1 July 1981 (preliminary).

15. Status Meeting (previously reported YAH-64 deficiencies and shortcomings), Hughes Helicopters, YAH-64 Program Manager, Development and Qualification Division AVRADCOM, USAAEFA, Culver City, CA, 5 August 1981.

APPENDIX B. DESCRIPTION

TABLE OF CONTENTS

GENERAL

1. The YAH-64 Advanced Attack Helicopter (fig. 1) is a tandem, two-place twin turbine engine, single main rotor aircraft manufactured by Hughes Helicopters (HH), a division of Summa Corporation. The main rotor is a four-bladed fully articulated system. It is supported by a stationary mast which transmits flight loads directly to the fuselage. The tail rotor is a four-bladed semi-rigid, delta-hinged system incorporating elastomeric teetering bearings. The rotors are driven by two General Electric YT 700-GE-700R engines through the power train shown in figure 2. An auxiliary power unit (APU) is installed primarily for starting the engines and to provide electrical and hydraulic power when the aircraft is on the ground and rotors are turning. The aircraft is designed to carry various combinations of ordinance stores internally in the ammunition bay and externally on the four wing store positions. The YAH-64 is designed to operate during day, night and marginal weather combat conditions using the Martin Marietta Target Acquisition Designation System (TADS)/Pilot's Night Vision System (PNVS). The test aircraft, S/N 77-23258, (photos 1 through 9) was configured with an aerodynamic mockup of the TADS/PNVS, 30mm chain gun and a HELLFIRE missile launcher loaded with four dummy missiles on each of the two inboard wing pylons. An alternate configuration included a HELLFIRE missile launcher loaded with four dummy missiles on all four wing pylons. The major modifications since Engineering Design Test 4 (EDT-4) consist of a 13 percent reduction in the pitch axis command augmentation system (CAS) authority and a linear reduction of the pitch axis CAS authority with airspeed from 100 percent at 80 knots true airspeed (KTAS) to zero percent at 280 KTAS. An additional change was the incorporation of an NOE/Approach mode of automatic stabilator functioning and the removal of the vertical vibration absorber. Various changes in external configuration which affect drag were made since EDT-4 and are reflected in table 1, and photos 10 through 12. Neither the Back Up Control System (BUCS) nor the Electronic Attitude Direction Indicator were operational during this test.

DIMENSIONS AND GENERAL DATA

	A&FC Part 1
Main Rotor	
Diameter (ft)	48
Blade chord (in.)	21.0*
Main rotor total blade area (ft^2)	166.5
Main rotor disc area (ft^2)	1809.56
Main rotor solidity (thrust weighted, no tip loss)	0.092
Airfoil	HH-02**
Twist	-9 deg
Number of blades	4
Rotor speed at 100 percent N_R (RPM)	289.3
Tip speed at 100 percent N_R (ft/sec)	727.09

*Includes tips
**Outer 20 inches swept 20 degrees and transitioned to an NACA 006 airfoil

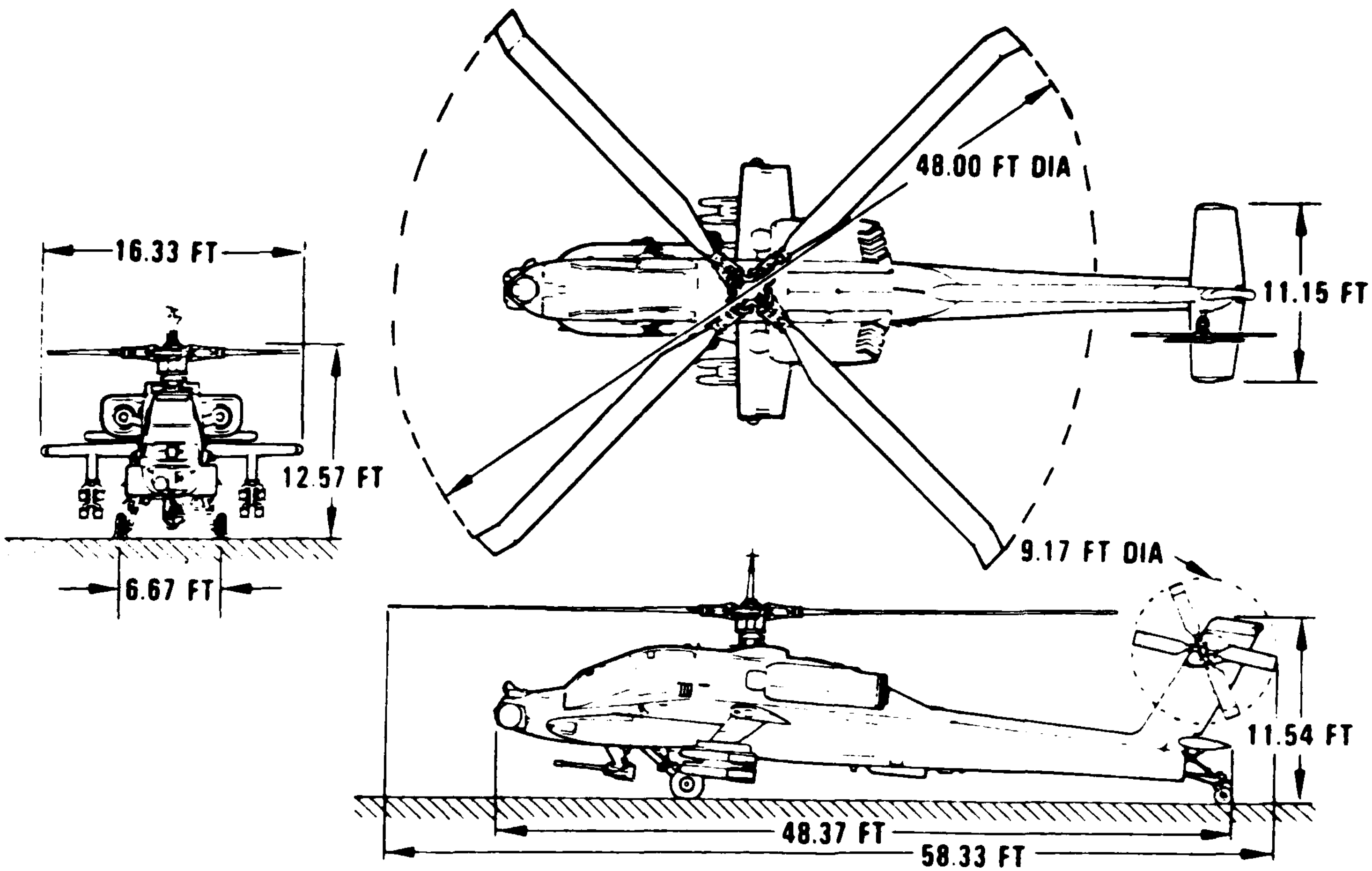


Figure 1. Aircraft Dimensions

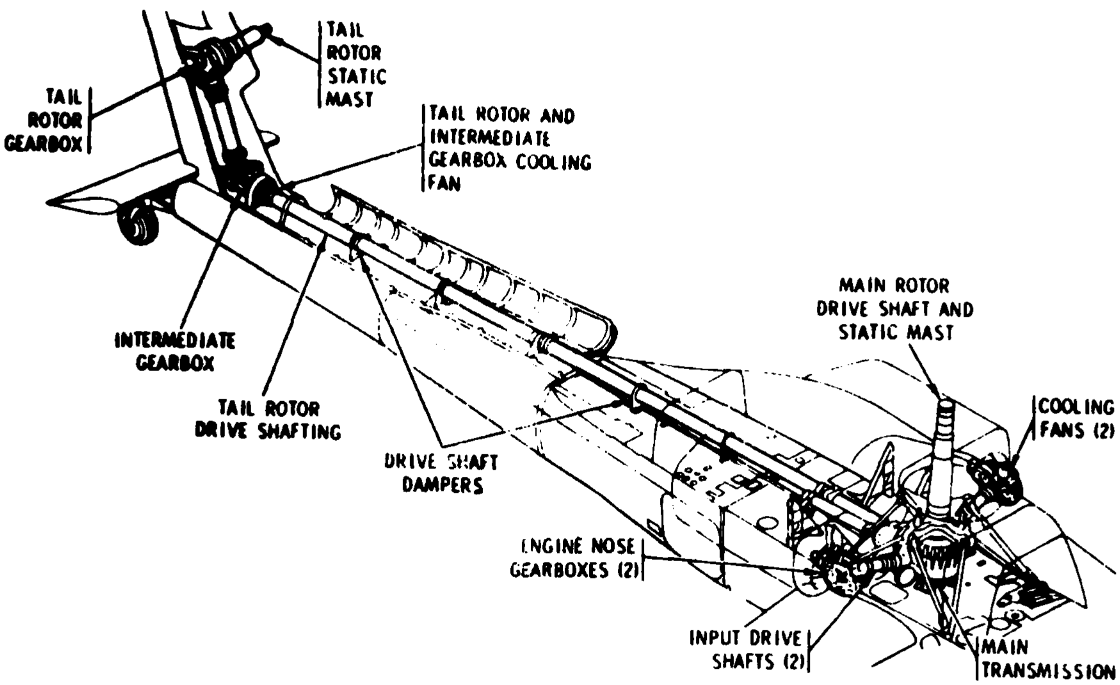


Figure 2. Powertrain

Photo 1. Front View (16 HELLFIRE Configuration)

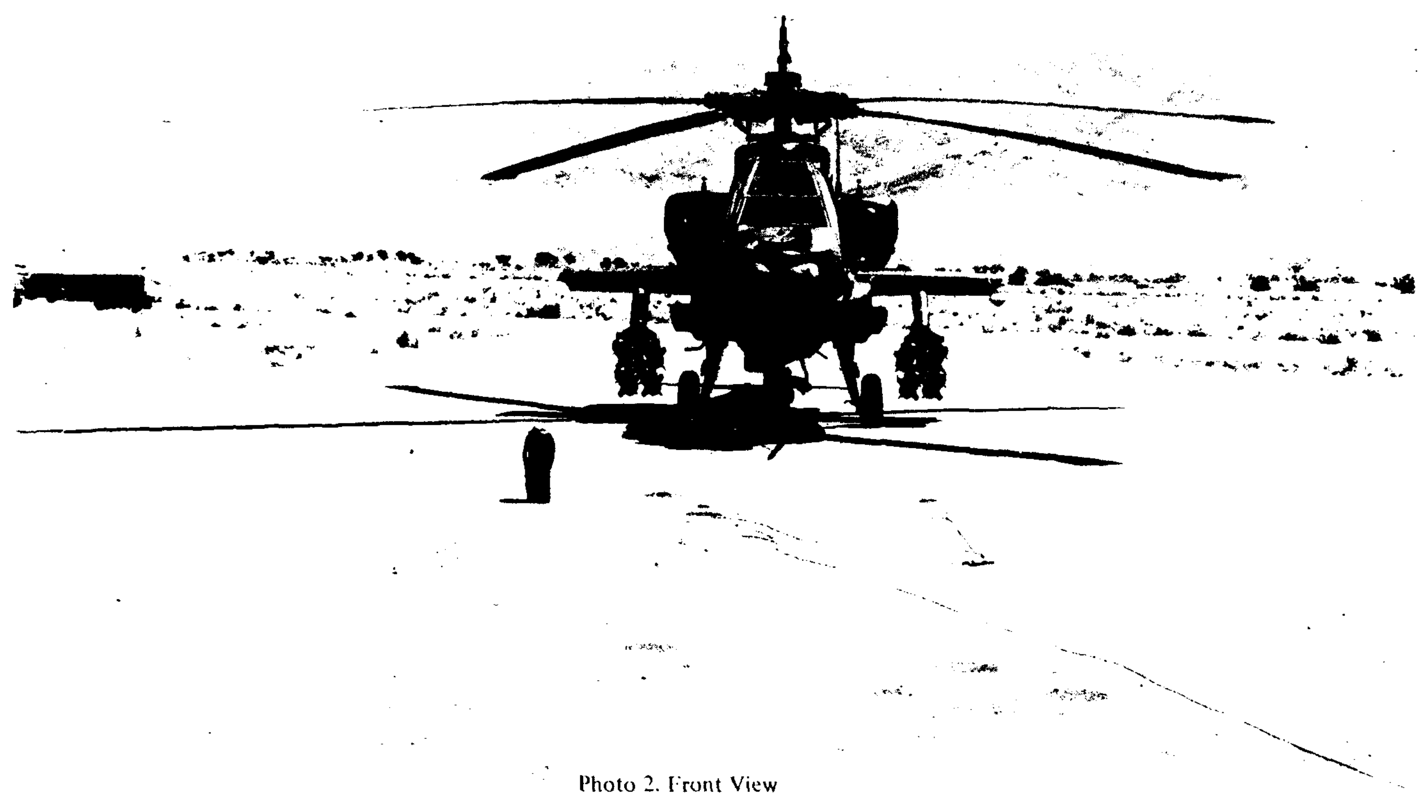


Photo 2. Front View

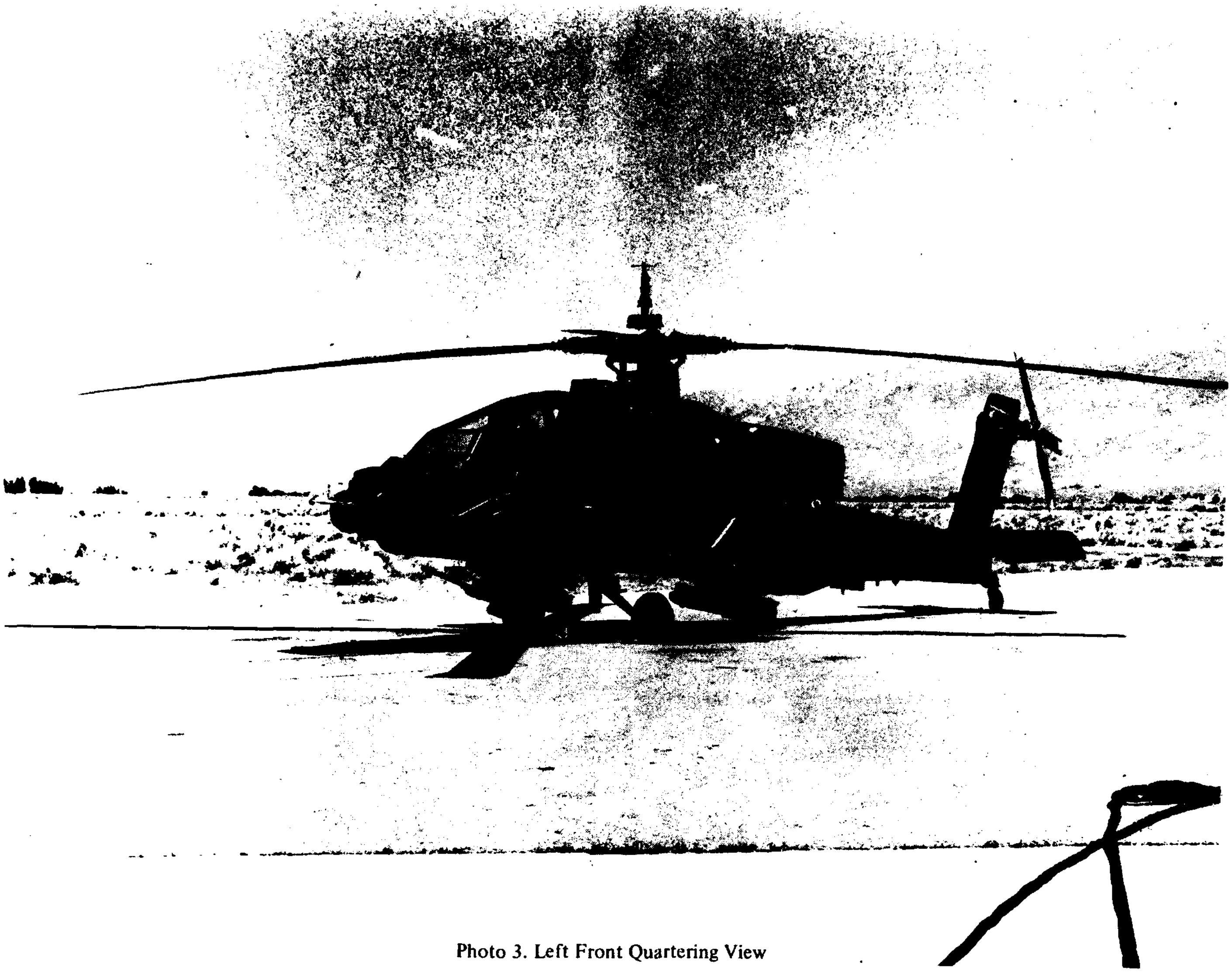


Photo 3. Left Front Quartering View

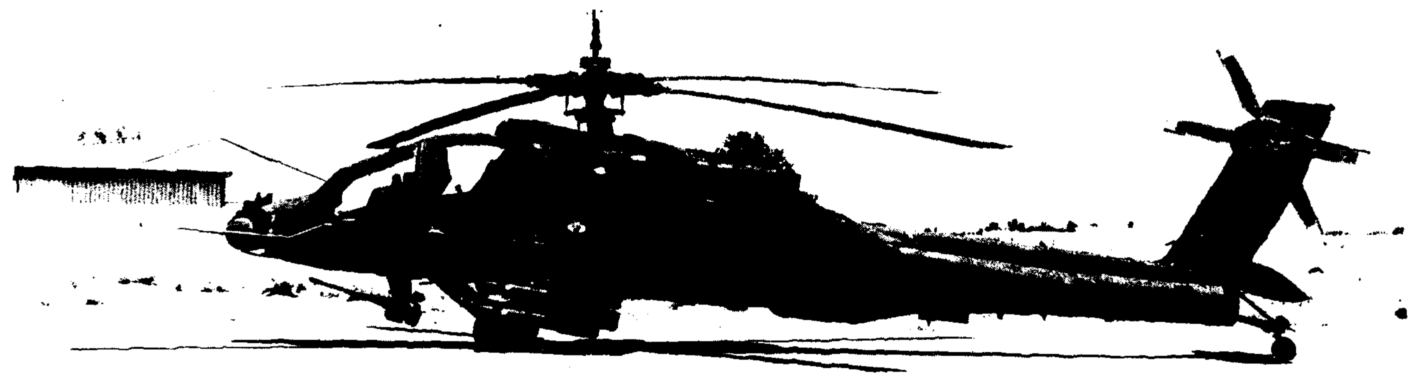


Photo 4. Left View

Photo 5. Left Rear Quartering View

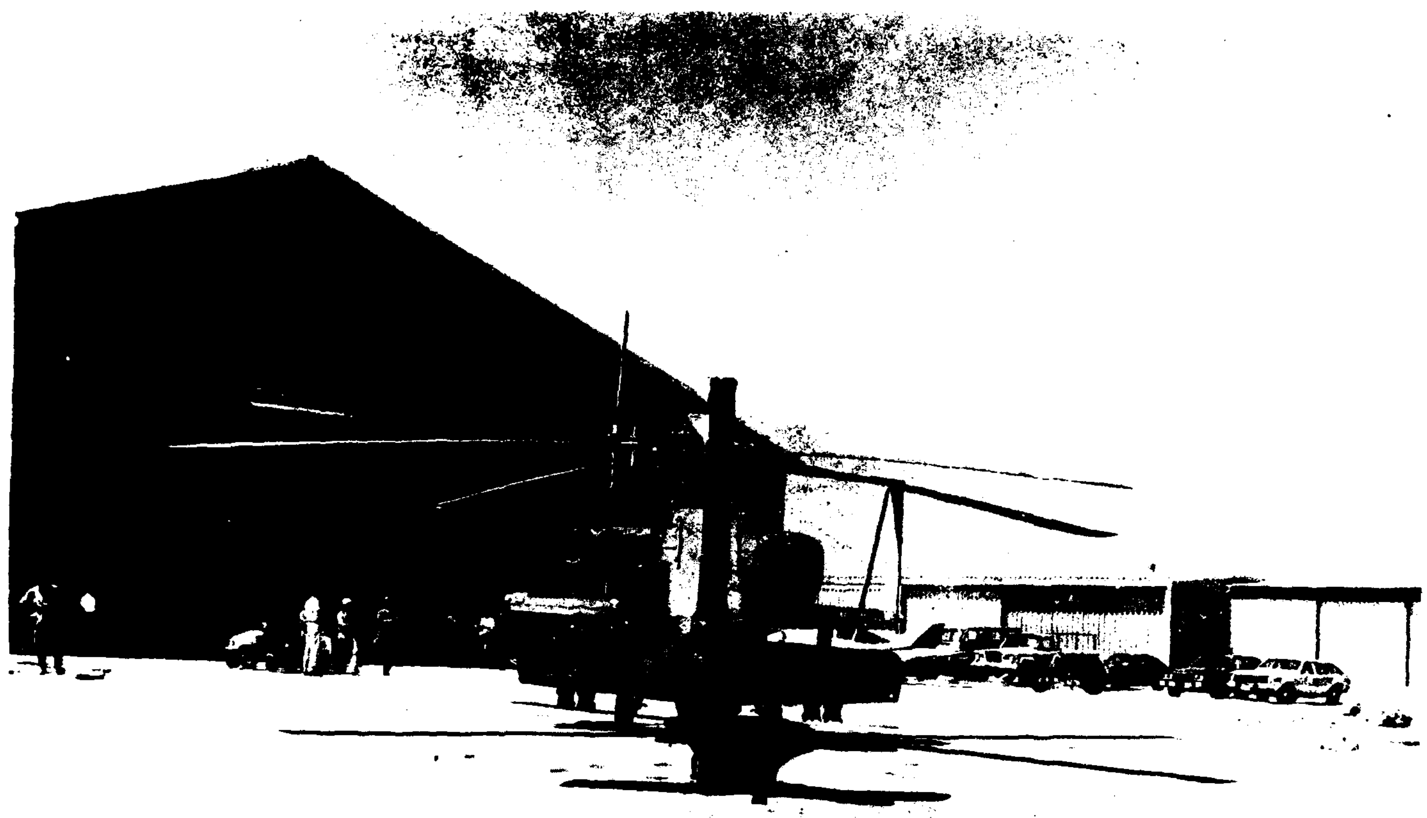


Photo 6. Rear View

Photo 7. Right Rear Quartering View

Photo 8. Right View

Photo 9. Right Front Quartering View

Table 1. External Configuration Changes Since EDT-4

1. Removal of wing tip light fairing, Engineering Change Request and Record (ECRR) Number 39322, dated 7 August 1980

2. Installation of one instrumented main rotor blade and two instrumented tail rotor blades

3. Installation of rotating PCM Canister (photo 10)

4. Installation of rotating PCM antenna brackets (photo 10)

5. Installation of one instrumented main rotor pitch change link (photo 10)

6. Installation of tail boom stiffeners (photo 11)

7. Installation of wing stiffening (photo 12).

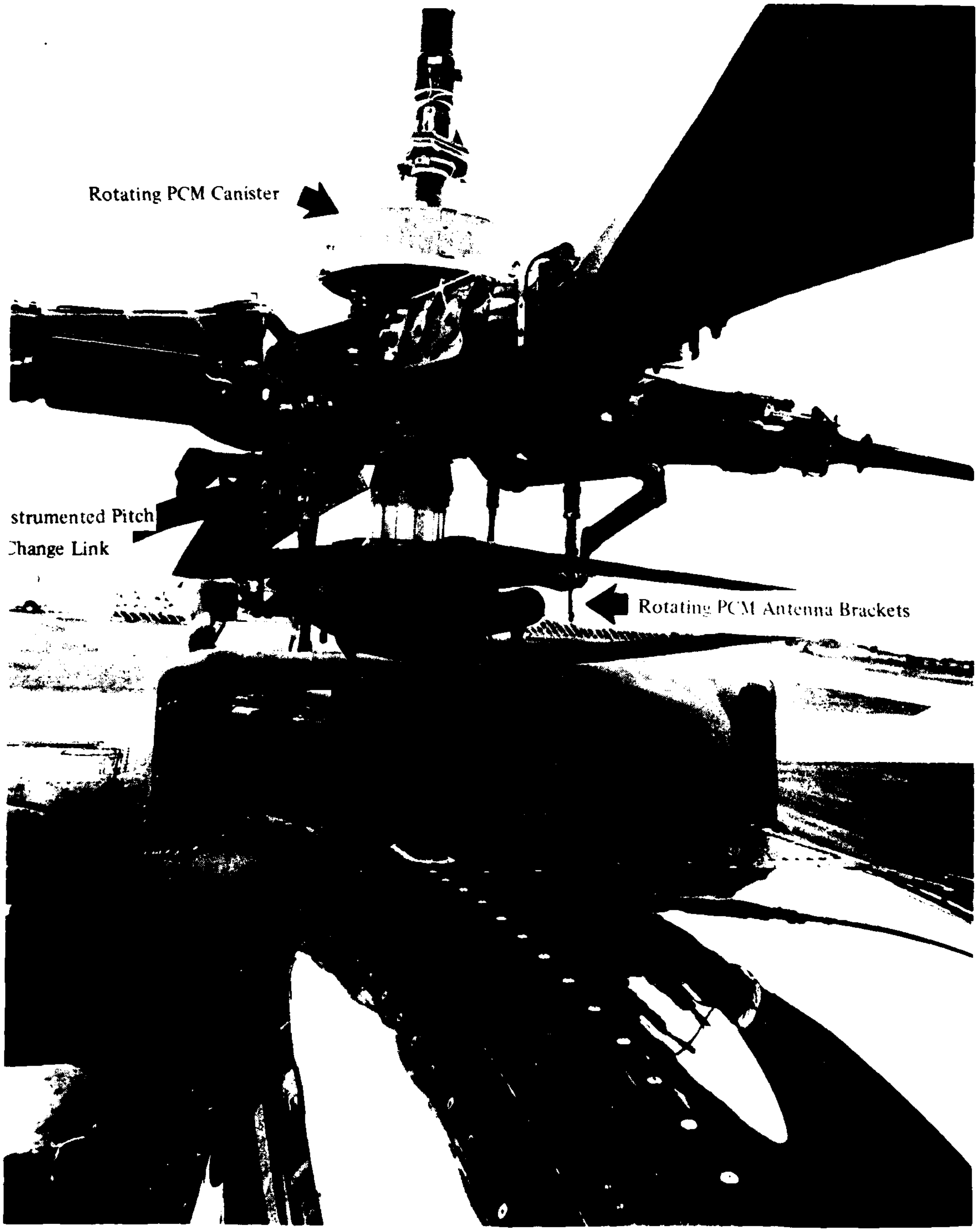


Photo 10. Main Rotor Rotating Instrumentation

Photo 11. Tailboom Stiffeners

Photo 12 Wing Stiffening

	A&FC Part 1
Tail Rotor	
Diameter (ft)	9.17
Chord constant (in.)	10
Tail rotor total blade area (ft^2)	14.89
Tail rotor disc area (ft^2)	66.0
Tail rotor solidity	0.2256
Airfoil	NACA 632-414 (modified)
Twist (deg)	8.8 washout
Number of blades	4
Rotor speed at 100 percent N_R (RPM)	1403.4
Distance from main rotor mast centerline (C_L) (ft)	29.67
Tip speed at 100 percent N_R (ft/sec)	673
Teetering angle (deg)	35
Horizontal Stabilizer/Stabilator	
Weight (lb)	77.3
Area (ft^2)	33.36
Span (ft)	10.67
Tip chord (ft)	2.65
Root chord (ft)	3.60
Airfoil	NACA 0018
Geometric aspect ratio	3.41
Incidence of chord line (deg)	Variable (45 degree leading edge up to 10 degree leading edge down).
Sweepback of leading edge (deg)	2.89
Sweepback of trailing edge (deg)	-7.23 deg (swept forward)
Dihedral (deg)	0
Vertical Stabilizer	
Area (from boom C_L) (ft^2)	32.2
Span (from boom C_L) (in.)	113.0
Root chord (at boom C_L) (in.)	44.0
Geometric aspect ratio	2.5
Airfoil	NACA 4415 (modified)
Leading edge sweep (deg)	29.4
Vertical stabilizer trailing edge deflection	16 deg left above W.L.196.0
Wing	
Span (ft)	16.33
Mean aerodynamic chord (in.)	45.9
Total area (ft^2)	61.59
Flap area (ft^2)	8.71 (fixed)
Airfoil at root	NACA 4418

Aircraft

Fuel quantity (gals.)	369
Design gross weight (lbs)	14525
Maximum gross weight (lbs)	17850

FLIGHT CONTROL DESCRIPTION

General

2. The YAH-64 helicopter employs a singe hydromechanical irreversible flight control system. The hydromechanical system is mechanically activated with conventional cyclic, collective and directional pedal controls, through a series of push-pull tubes attached to four airframe-mounted hydraulic servoactuators. The four hydraulic servoactuators control longitudinal cyclic, lateral cyclic, main rotor collective and tail rotor collective pitch. Cyclic and directional servoactuators incorporate integral SAS actuators. Hydraulic power is supplied by two independent 3000-psi hydraulic systems which are powered by hydraulic pumps mounted on the accessory gearbox to allow full operation under a dual-engine failure condition. A Digital Automatic Stabilization Equipment (DASE) system is installed to provide rate damping. The DASE control authority is limited to 10 percent of pilot control authority in pitch, roll, and yaw. The DASE also provides attitude hold and a Hover Augmentation System. An electrically-actuated horizontal stabilator is attached to the lower aft side of the vertical stabilator. Movement of the stabilator can be controlled either manually or automatically. A Trim Feel System (TFS) is incorporated in the cyclic and pedal controls to provide a control force gradient with control displacement from a selected trim position. A trim release switch, located on the cyclic grip, provides either a momentary or continuous interruption of the TFS in all axes simultaneously to allow the cyclic or pedal controls to be placed in a new trim position. Full control travel is 11.0 inches in the longitudinal control, 9.0 inches in the lateral control, 12.8 inches in the collective control, and 4.7 inches in the directional pedals.

Cyclic Control System

3. The cyclic control system (fig. 3) consists of dual-tandem cyclic controls attached to individual support assemblies in each cockpit. The support assembly houses the primary longitudinal and lateral control stops, and two linear variable displacement tranducers (LVDT) designed to measure electrically the longitudinal and lateral motions of the cyclic for DASE computer inputs. A series of push-pull tubes and bellcranks transmits the motion of the cyclic control to the servoactuators and the mixer assembly. Motion of the mixer assembly positions the nonrotating swashplate, which transmits the control inputs to the rotating swashplate to control the main rotor blades in cyclic pitch (fig. 4). The cyclic stick grips are shown in figure 5. A stick fold linkage is provided to allow the copilot/gunner (CPG) to lower the cyclic stick to prevent interference when operating the weapon systems.

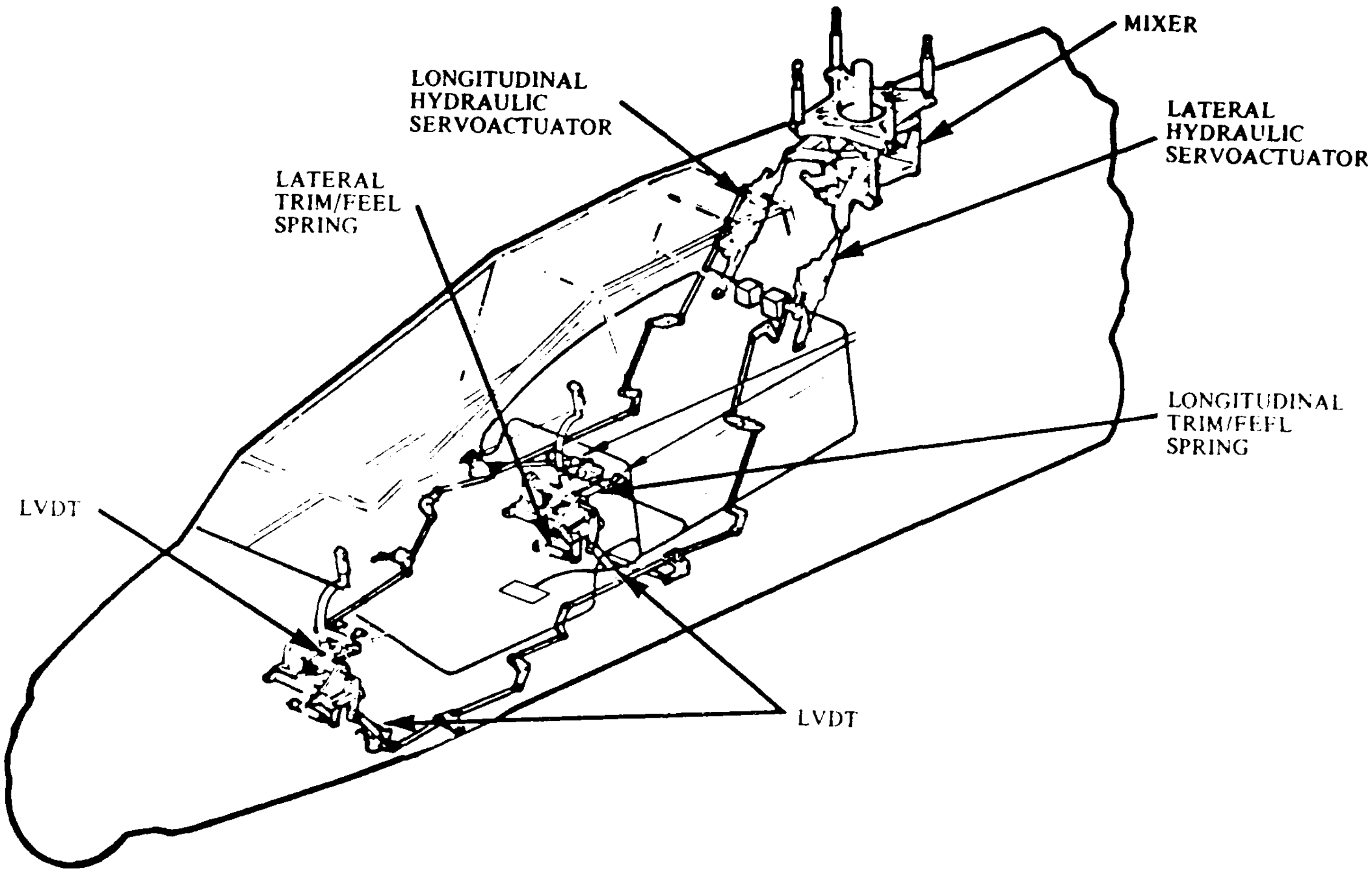


Figure 3. Cyclic Control System

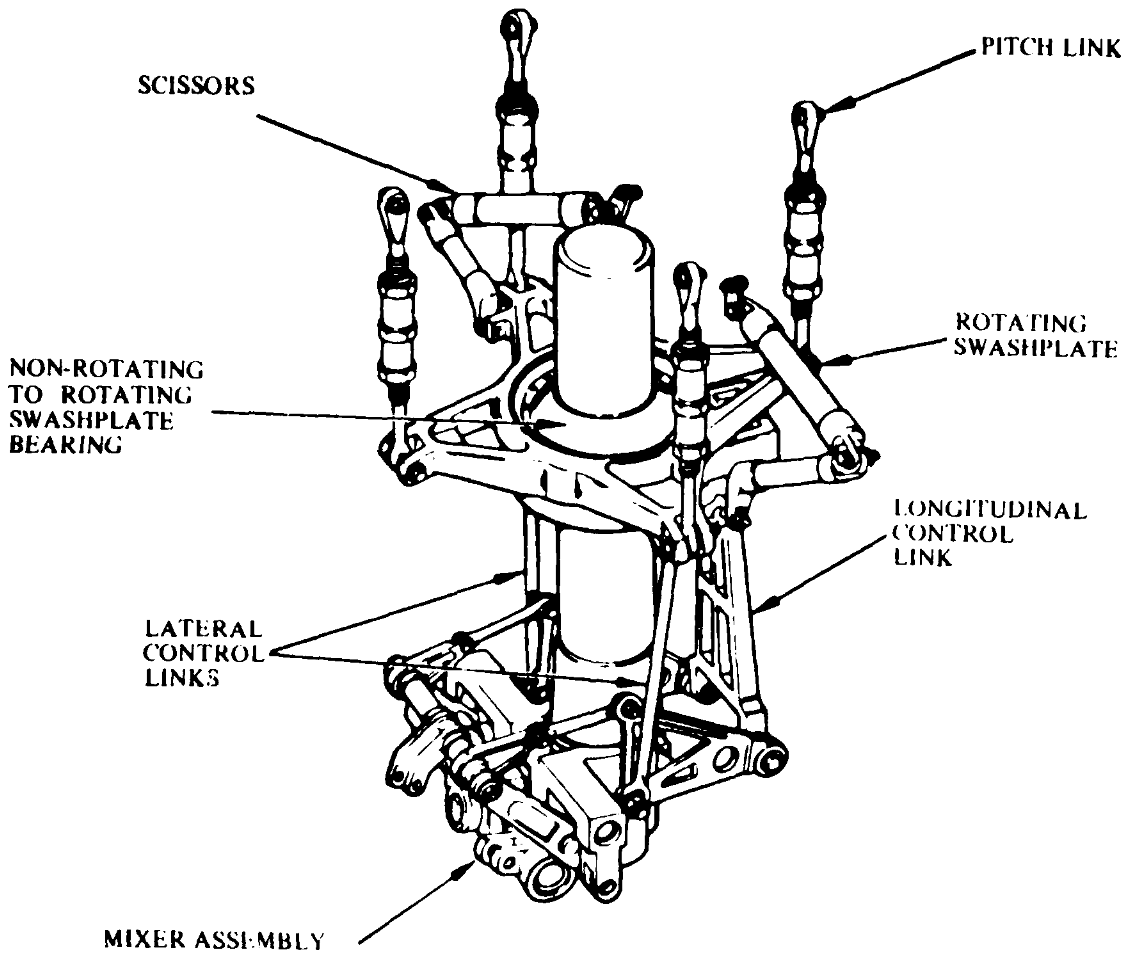
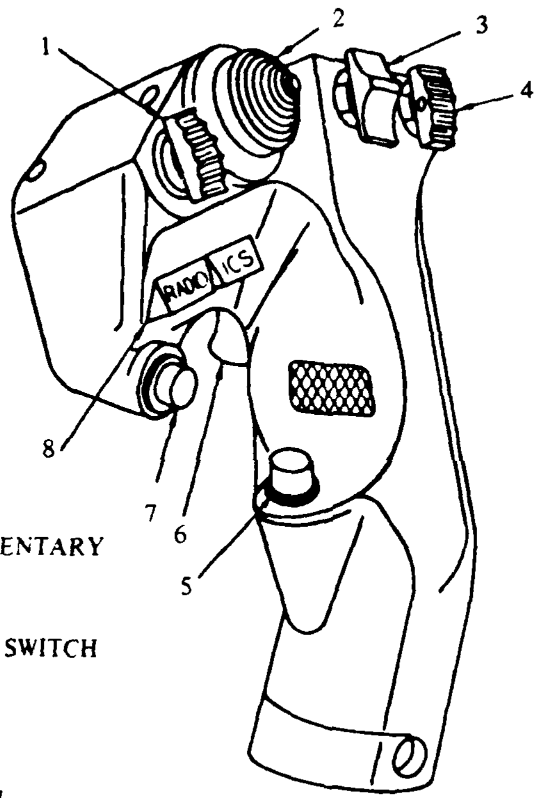
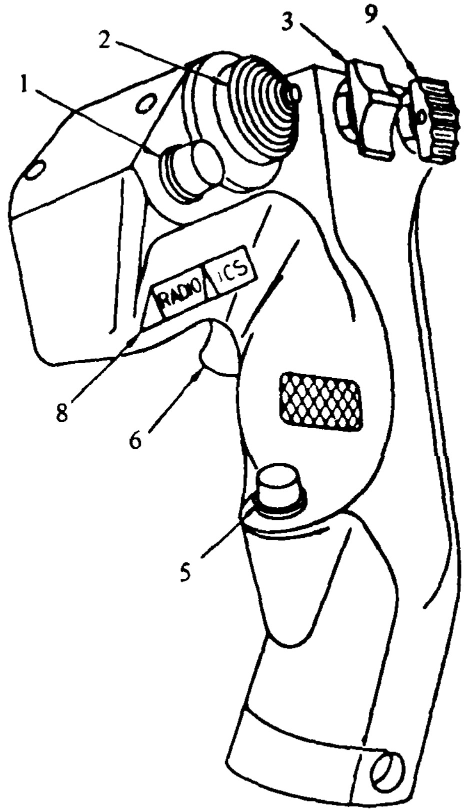


Figure 4 Main Rotor Swashplate Assembly

PILOT CYCLIC STICK GRIP



COPILOT/GUNNER CYCLIC STICK GRIP



1. TRIM RELEASE SWITCH
   CPG GRIP-MOMENTARY
   PILOT GRIP-DISCRETE/MOMENTARY

2. WEAPONS ACTION SWITCH

3. FLIGHT MODE SYMBOLOGY SWITCH

4. TO BE DETERMINED

5. DASE RELEASE SWITCH

6. GUARDED TRIGGER SWITCH

7. REMOTE TRANSMITTER SELECTOR SWITCH (PILOT GRIP ONLY)

8. RADIO, ICS ROCKER SWITCH

9. NEAR/FAR FOCUS

Figure 5. Cyclic Stick Grips

Collective Control System

4. The collective pitch control subsystem (fig. 6) consists of dual-tandem controls which transmit collective control inputs to the main rotor through a series of push-pull tubes and bellcranks attached to the collective servoactuator. Motion of the servoactuator is transmitted through the mixer assembly to the swashplate to control the main rotor blades in collective pitch. Collective inputs are also transmitted to the load demand spindle of each engine hydromechanical unit (HMU). The HMU meters the fuel as appropriate to provide collective pitch compensation. Located at each collective control base assembly are the primary control stop, and LVDT, and a 1 g balance spring. The LVDT supplies electical inputs to the stabilator control units.

5. The collective control stick (fig. 7) incorporates a switch box assembly, an engine chop collar, a stabilator control panel and an adjustable friction control. The engine chop collar allows rapid deceleration of the engine to flight idle, primarily to allow immediate action in the event of a tail rotor failure.

Directional Control System

6. The directional control system (fig. 8) consists of a series of push-pull tubes and bellcranks which transmits directional pedal inputs to the tail rotor hydraulic servoactuator located in the vertical stabilizer. Attached to each directional pedal assembly are the primary tail rotor control stops and one LVDT. A mechanical hardstop was installed on the test aircraft directional pedal to limit tail rotor blade angle to 27 degrees. However, this hardstop did not prevent actuator overtravel (beyond 27 degree blade angle) with DASE (command augmentation system) input. Two sets of wheel brake cyclinders are attached to the directional pedals and a 360 degrees swiveling tail wheel is incorporated. The tail wheel may be locked in the trailing position by means of a switch located on the pilot's instrument panel.

Trim Feel System

7. A TFS is incorporated in the longitudinal, lateral, and directional control systems. The TFS uses individual magnetic brake clutch assemblies in each of the control linkages. Trim feel springs are incorporated to provide a control force gradient and positive control centering. The electromagnetic brake clutch is powered by 28 VDC and is protected by the trim circuit breaker. A complete DC electrical failure will disable the TFS and allow the cyclic and directional pedals to move freely without resistance from the trim feel springs. The trim release switch on the pilot's cyclic grip allows either momentary or discreet release of the TFS. The CPG has a momentary release capability only.

Horizontal Stabilator

8. The horizontal stabilator is attached to the aft lower portion of the vertical stabilizer. A dual, series 28 VDC electromechanical actuator allows incidence changes of +45 to -10 degrees leading edge up (LEU) of travel. Safety features include an automatic shutdown capability which allows operation in the manual mode by means of a stabilator control panel located on each collective stick. An audio tone is associated with the failure of the automatic mode of operation. A stabilator kill switch, located on the pilot's collective stick, disables both the automatic and manual operation to protect against a hardover failure. There are

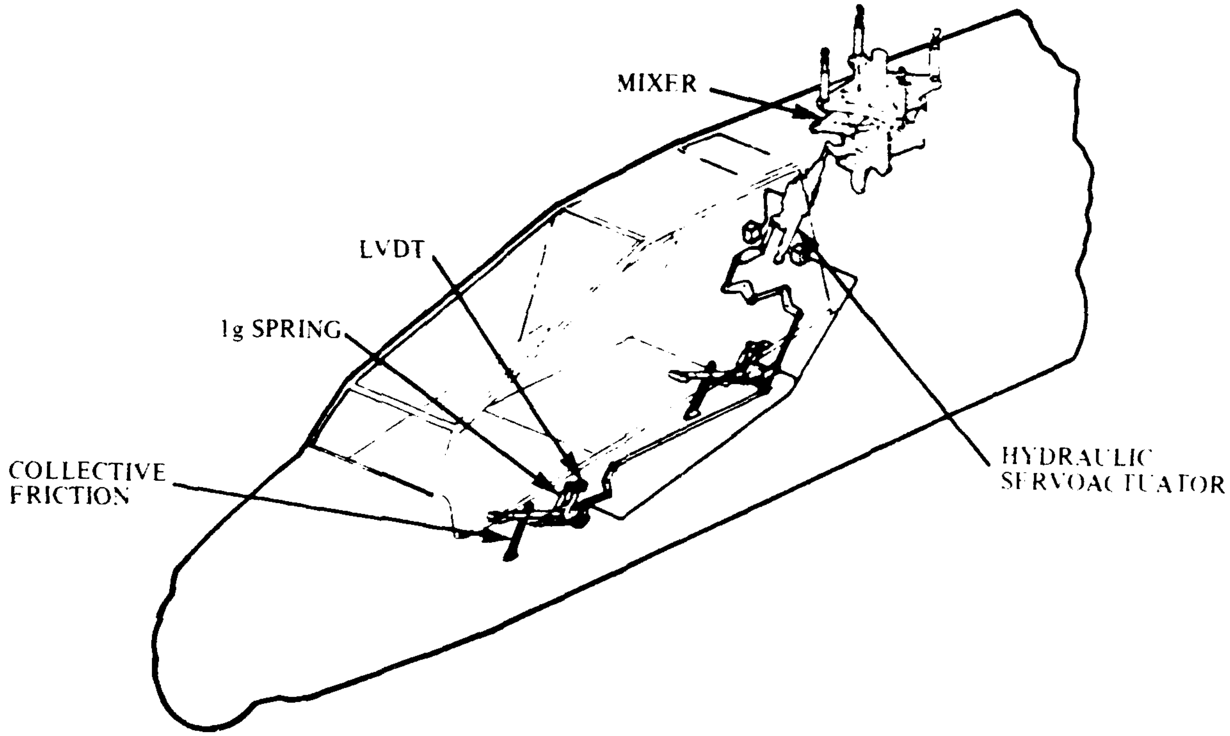


Figure 6. Collective Control Subsystem

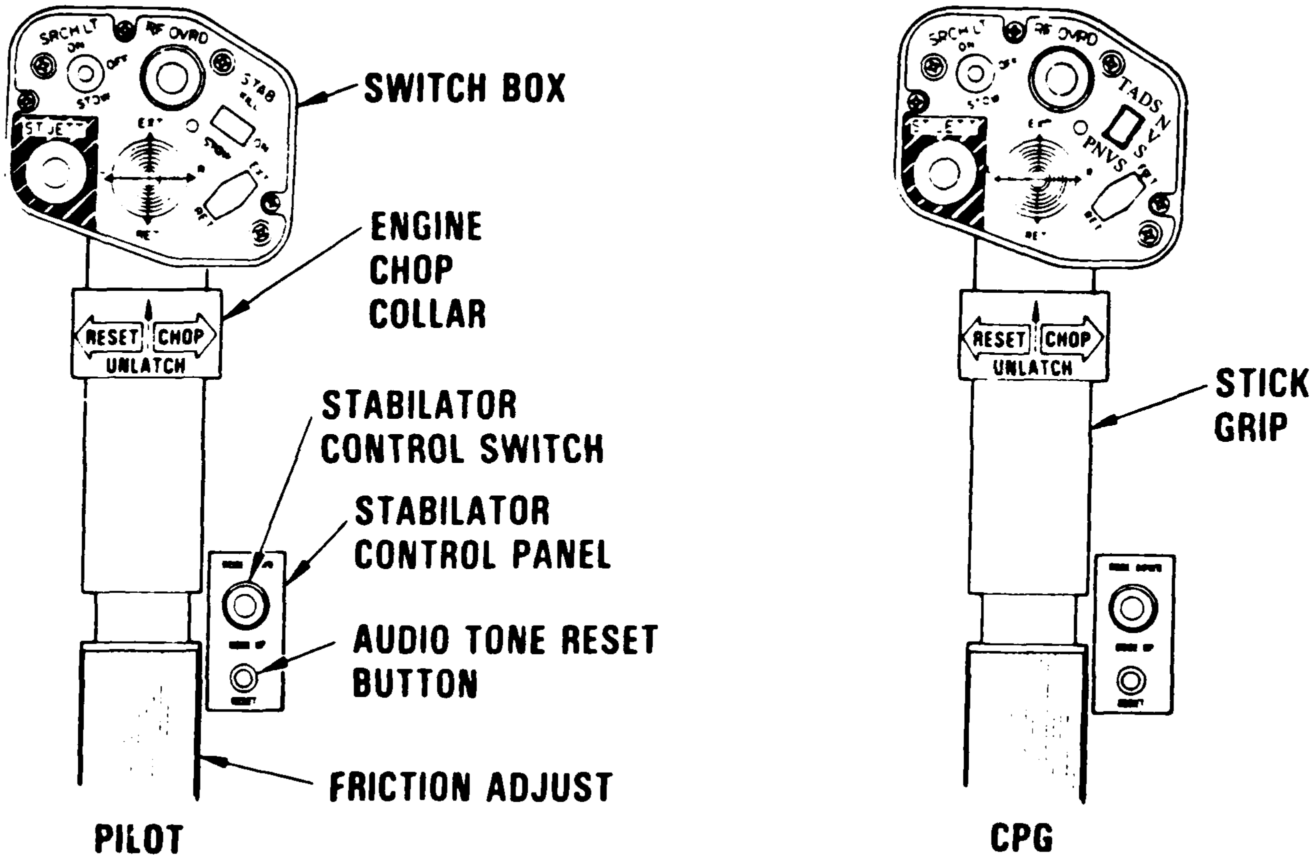


Figure 7. Collective Stick and Switch Box

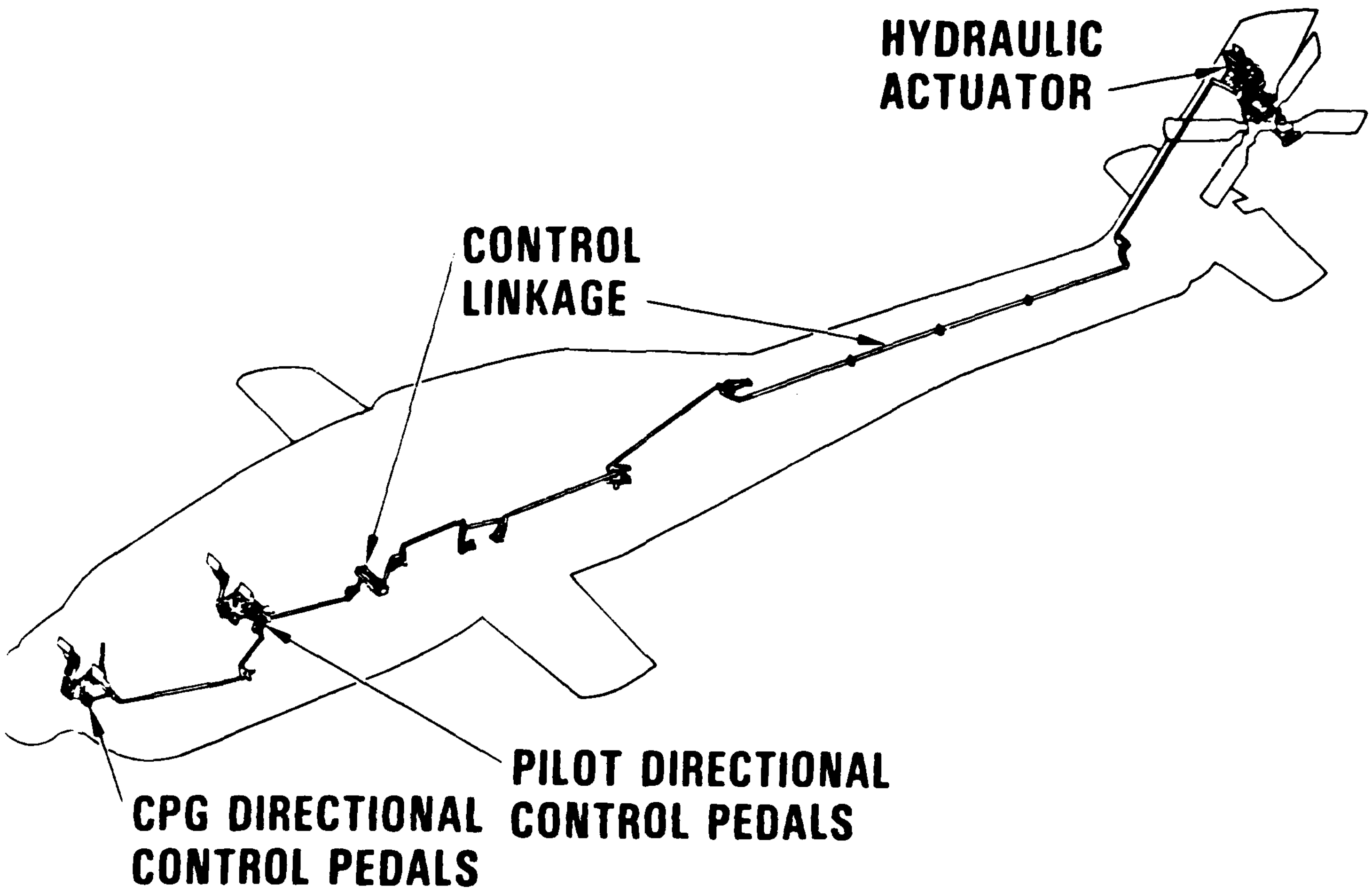


Figure 8. Directional Controls

three modes of stabilator operation: the automatic mode, the NOE/Approach mode and the manual mode. The stabilator is controlled in the automatic mode by two stabilator control units (SCU's). Each SCU controls one side of the dual actuator. Both SCU's receive collective control position information from an LVDT. Two independent pitch rate gyros provide pitch rate information to the SCU's (one gyro for each SCU). The Air Data System (ADS) provides airspeed to both SCU's. Additionally, the left-handed pitot-static system supplies airspeed to one SCU and the right-hand system provides airspeed to the other SCU. Both SCU's receive position information from both sides of the dual actuator. The maximum rate of stabilator travel is 7 degrees per second.

9. The automatic mode is operational when the aircraft has normal AC and DC electrical power applied. Automatic positioning of the stabilator during flight is primarily a function of airspeed and collective position as shown in figure 9. The stabilator also responds with a low gain (0.2 deg/sec/sec) and limited authority (±5.0 deg) to pitch rate inputs to the SCU. Software limits in the SCU limit the incidence change in the automatic mode to those shown in figure 9.

10. The NOE/Approach Mode is selected through the NOE/APPR mode switch on the pilot's DASE panel and will stay engaged at any speed. The mode becomes operational below 80 knots indicated airspeed (KIAS) and will bias the stabilator to 25 degrees LEU at 3.6 deg/sec stabilator rate. The mode can be disengaged by manual mode selection below 80 knots, or activation of the DASE release or AUTO STAB reset switch. Acceleration through 80 KIAS will automatically engage the normal automatic schedule and the stabilator will move at 3.6 deg/sec for the first 10 seconds. Failure to revert to automatic schedule will result in system disengagement with visual and aural indications.

11. The manual mode can be selected below 80 KIAS through pilot's and CPG's manual control switch on the collective stick. Manual control selection will result in STAB FAIL caution and warning annunciation. Selection of automatic mode can be accomplished by pressing the AUTO STAB reset switch on the pilot's or CPG's collective stick. The stabilator will move at 7 deg/sec rate to the automatic mode schedule position. Acceleration through 80 KIAS in manual mode will automatically engage the automatic mode and the stabilator will move at 7 deg/sec.

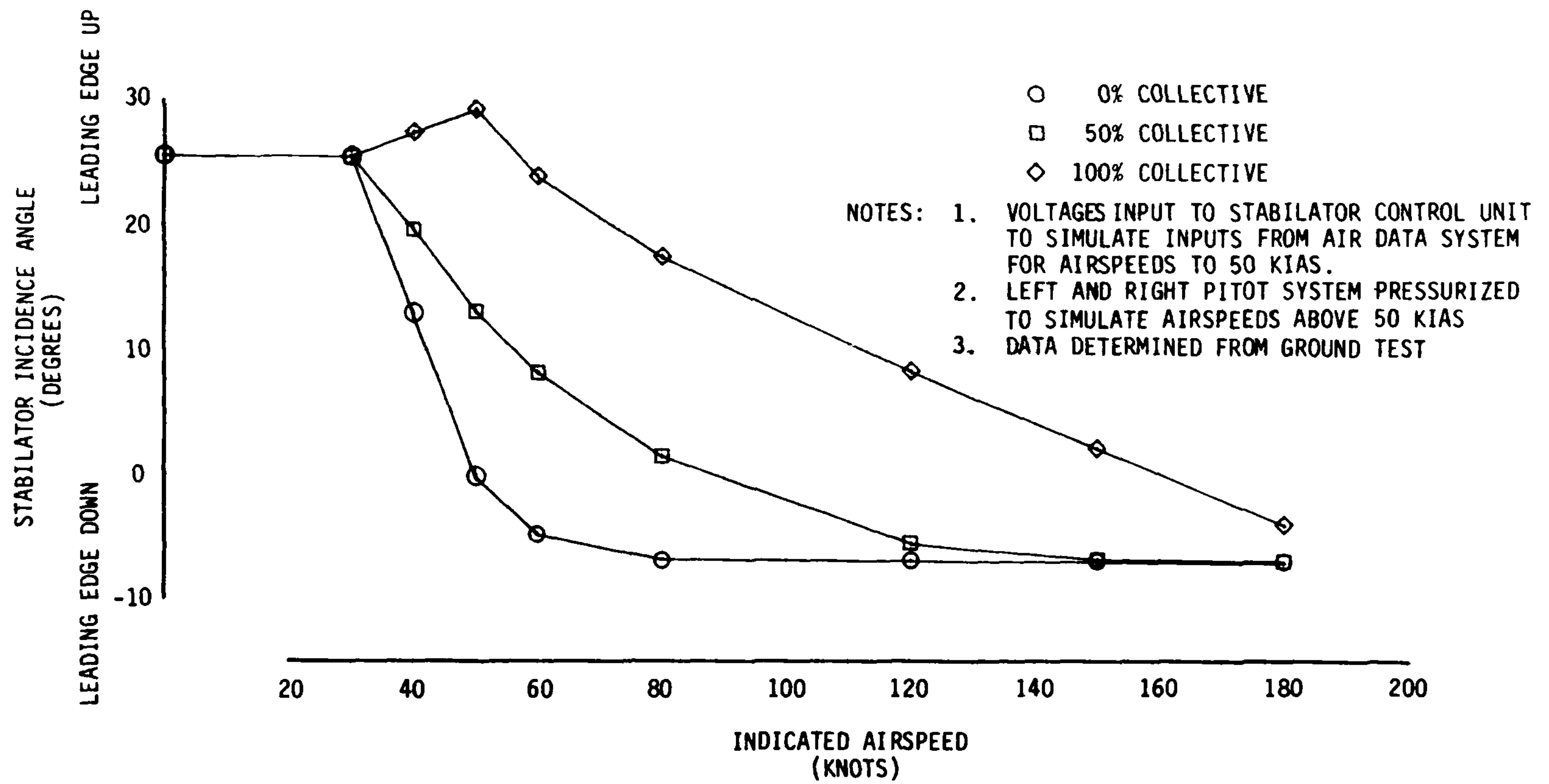
12. The SCU's have a fault detection feature which will switch the stabilator mode of operation from automatic to manual if any of the following conditions are sensed:

a. A mismatch between the positions of the two halves of the actuator equivalent to 10 degrees of stabilator travel (If there is a runaway failure of one side of the actuator, this feature will disable the automatic mode after 10 degrees of stabilator travel.)

b. A mismatch in the rates of actuator travel of more than 10 degrees per second (This could only occur if one side of the actuator were extending while the other was retracting.)

c. The stabilator at a position of 20 degrees or greater with an airspeed greater than 110 KIAS

FIGURE 9
STABILATOR SCHEDULE
YAH-64 USA S/N 77-23258



d. A stabilator angle of 25 degrees or greater with airspeed less than 80 KIAS

e. Improper AC voltage (less than 23 VAC for more than 0.15 seconds) or improper current (greater than 6.5 amps DC for more than 5.5 seconds).

Flight Control Rigging

13. A flight controls rigging check was performed in accordance with procedures outlined in HH Experimental Test Procedure (ETP) 7-211500000, dated 1 December 1980, (main and tail rotor controls) and ETP 7-211123600, dated 21 April 1980 (horizontal stabilator). Horizontal stabilator schedule is shown in figure 9. Tables 2 and 3 present the collective and cyclic rigging.

14. Tail rotor rigging is shown below:

Full right pedal: 15.1 degrees thrust to left
Full left pedal: 27.0 degrees thrust to right

Digital Automatic Stabilization Equipment

15. The DASE provides rate damping (SAS), control augmentation (CAS), hover augmentation (HAS), attitude hold, and turn coordination. A preliminary DASE description is provided in reference 14, appendix A. The DASE is controlled by the digital automatic stabilization equipment computer (DASEC). The DASEC receives information from several sources. The heading and attitude reference system (HARS) provides the DASEC with aircraft angular velocities (3 axes), aircraft attitudes (pitch and roll), and inertial horizontal velocity (measured by the Doppler radar). The Air Data System (ADS) provides lateral and longitudinal airspeed, and sideslip angle. The LVDT's provide longitudinal, lateral, and directional control position information. The electronic attitude direction indicator (EADI) provides turn rate. The DASEC processes this information and commands control inputs through the electro-hydraulic servo valves on the longitudinal, lateral, and directional servoactuators. The DASE authority is limited in each axis to ±10 percent of the pilot's control authority. Figure 10 presents a block diagram of the DASE.

16. The SAS function of the DASE system provides rate damping in pitch, roll, and yaw axes. Each axis is separately engageable through a magnetically held toggle switch on the DASE panel shown in figure 11. The CAS is used to augment the pilot control inputs. CAS is an automatic function of the DASE whenever pitch and roll SAS are selected and yaw SAS is selected below 60 KTAS.

17. A limited authority HAS mode is provided through pitch and roll SCAS using rates, attitudes and Doppler corrected inertial velocities from the HARS. HAS is used to reduce pilot workload in gusty conditions by assisting the pilot in maintaining a desired hover position. HAS is engageable through the DASE panel using the same switch designated for attitude hold below 25 knots ground speed and 50 KTAS whenever pitch and roll SCAS are engaged. Additionally, a heading hold mode is provided through the yaw SCAS using aircraft heading information from the HARS. This function is engageable whenever yaw SCAS is engaged and the HAS switch is selected.

Table 2. Angle Measurements
Pilots Collective and Cyclic Controls

Blade Azimuth Position (deg)	Item	Rig Pins: Collective	Rig Pins: Longitudinal Cyclic	Rig Pins: Lateral Cyclic	Stick Position: Collective	Stick Position: Longitudinal	Stick Position: Lateral	Measured Clinometer Angle (deg)	Leading Edge Up or Down
$\psi = 90$	1	In	In	In	Rig	Rig	Rig	0.2	Down
	2	In	Out	In	Rig	Fwd	Rig	21.0	Down
	3	In	Out	In	Rig	Aft	Rig	11.4	Up
	4	In	In	In	Rig	Rig	Rig	0.4	Down
	5	Out	In	In	Up	Rig	Rig	9.5	Up
	6	Out	In	In	Down	Rig	Rig	9.7	Down
	7	In	In	In	Rig	Rig	Rig	0.4	Down
$\psi = 270$	8	In	In	In	Rig	Rig	Rig	0.4	Down
	9	In	Out	In	Rig	Fwd	Rig	20.8	Up
	10	In	Out	In	Rig	Aft	Rig	11.0	Down
	11	In	In	In	Rig	Rig	Rig	0.9	Up
$\psi = 0$	12	In	In	In	Rig	Rig	Rig	0.5	Down
	13	In	In	Out	Rig	Rig	Left	11.5	Up
	14	In	In	Out	Rig	Rig	Right	7.4	Down
	15	In	In	In	Rig	Rig	Rig	0.3	Down
$\psi = 180$	16	In	In	In	Rig	Rig	Rig	0.0	
	17	In	In	Out	Rig	Rig	Left	11.2	Down
	18	In	In	Out	Rig	Rig	Right	7.8	Up
	19	In	In	In	Rig	Rig	Rig	0.1	Down

Table 3. Computation of Blade Angle Travel Pilots Collective and Cyclic Controls

Computation	Travel (deg)	Tolerance (deg)
LONGITUDINAL CYCLIC		
1. Forward = 1/2 (Item* 9 - Item 2) = (If Item 2 is leading edge down add Item 2)	20.9	20° (minimum)
2. Aft = 1/2 (Item 3 - Item 10) = (If Item 10 is leading edge down add Item 10)	11.2	10° (minimum)
LATERAL CYCLIC		
3. Left = 1/2 (Item 13 - Item 17) = (If Item 17 is leading edge down add Item 17)	11.4	10.5° (minimum)
4. Right = 1/2 (Item 18 - Item 14) = (If Item 14 is leading edge down add Item 14)	7.6	7.0° (minimum)
COLLECTIVE		
5. Full pitch travel = (Item 5 - Item 6) = (If Item 6 is leading edge down add Item 6)	19.2	18.0° (minimum)
6. Collective pitch full down Measured @ 3/4 radius (Theo. chord line)		-1° to +2°
Measured @ pitch housing (Bolt pad machined surface 2.4 inches inboard of lead-lag hinge)	-9.7	-10° to -7°

*Item numbers obtained from table 2

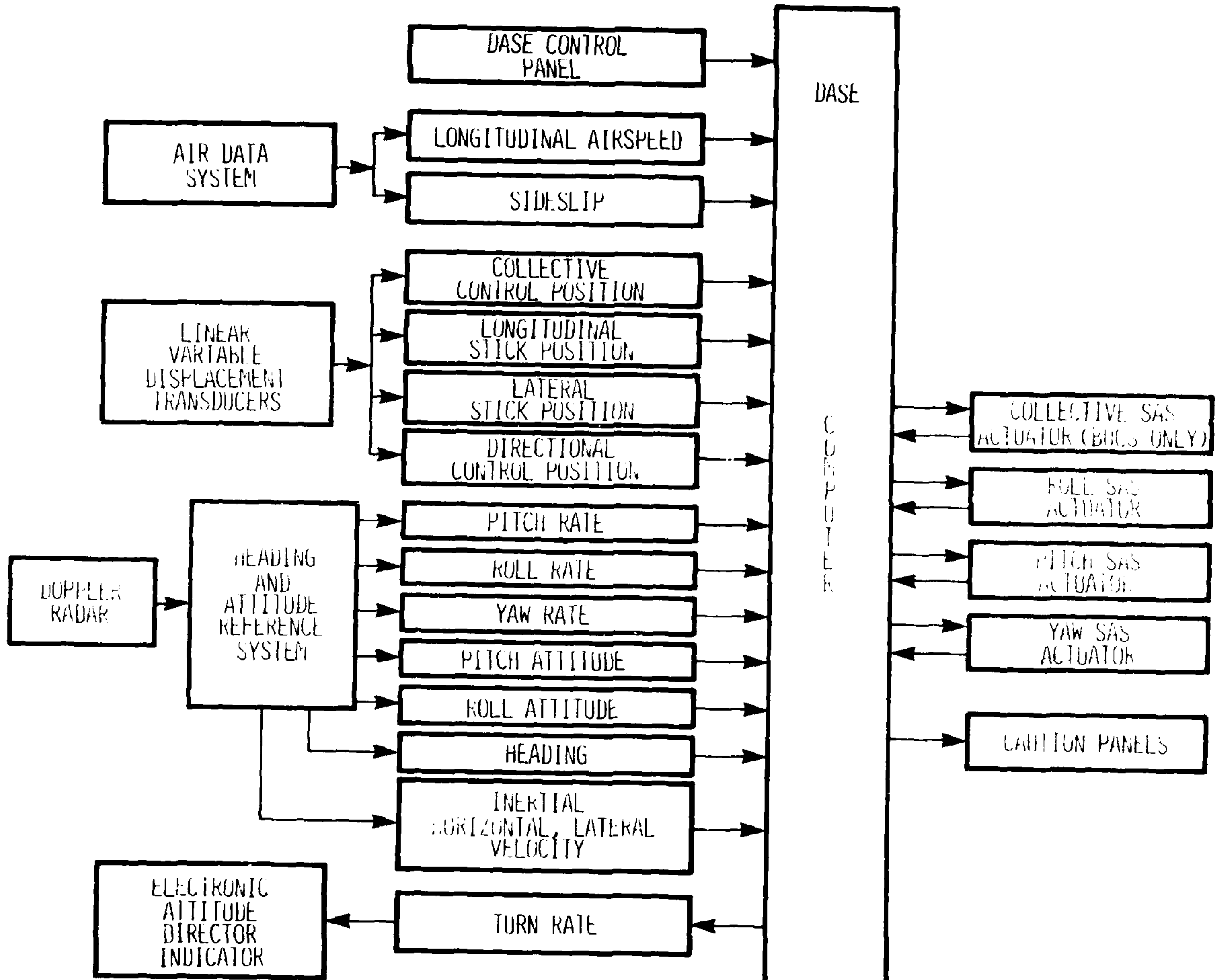


Figure 10. Digital Automatic Stabilization Equipment Block Diagram

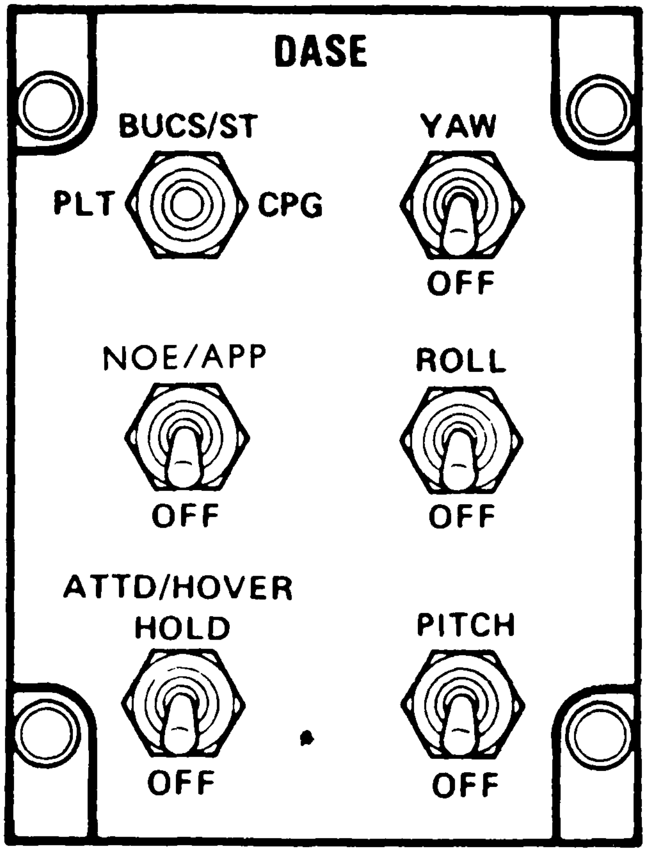


Figure 11. DASE Control Panel

18. A limited authority attitude hold mode is provided through pitch and roll SAS. Attitude Hold is engageable through the DASE panel above 60 KTAS whenever pitch and roll SAS are engaged. Attitude Hold will automatically disengage whenever the airspeed is decreased to 50 KTAS.

19. A limited authority turn coordination function is provided through yaw SAS using sideslip information from the ADS. This function is automatically provided above 60 KTAS whenever yaw SAS is engaged.

HYDRAULIC SYSTEM

General

20. The hydraulic system consists of four hydraulic servoactuators powered simultaneously by two independent 3000-psi hydraulic systems. Each servoactuator simultaneously receives pressure from the primary and utility systems to drive the dual-tandem actuators. This design allows the remaining system to automatically continue powering the servos in the event of a single hydraulic system failure. The two systems (primary and utility) are driven by the accessory gearbox utilizing variable displacement pumps, independent reservoirs and accumulators. The APU drives all accessories, including the hydraulic pumps, when the aircraft is on the ground and the rotor is not running. The accessory gearbox is driven by the main transmission during flight and provides for normal operation of both hydraulic systems during autorotation. An emergency hydraulic system is provided to allow emergency operation of the flight controls in the event of a dual system failure.

Primary Hydraulic System

21. The primary hydraulic system (fig. 12) consists of a one-pint capacity reservoir, which is pressurized to 30 psi using air from the shaft-driven compressor; an accumulator, which has a nitrogen precharge of 1600 psi, designed to reduce surges in the hydraulic system; and a primary manifold that directs the fluid to the lower side of the four servoactuators. The primary system also provides the hydraulic pressure for operation of the DASE and BUCS functions.

Utility Hydraulic System

22. The utility hydraulic system (fig. 13) consists of an air pressurized 1.3 gallon reservoir and a 3000- psi accumulator which drives the APU starting motor. The utility manifold directs fluid to the upper side of the servoactuators, the stores pylon system, tail wheel lock mechanism, area weapon turret drive, and rotor brake. Other manifold functions include an auxiliary isolation check valve which isolates the area weapon turret drive and external stores actuators when either a low pressure or low fluid condition exists; a low pressure sensor isolates the accumulator as an emergency hydraulic source for the servoactuators in the event of a dual hydraulic system failure. The accumulator assembly stores enough fluid for emergency operation of the flight controls through four full strokes of the collective stick and one 180 degrees heading change. The emergency system may be activated by either the pilot's or CPG's emergency switch. An electrically activated emergency shutoff valve is designed to isolate the utility side of the directional servoactuator and the tail wheel lock mechanism when a low fluid condition exists. However, the electrical connections were not installed during this test.

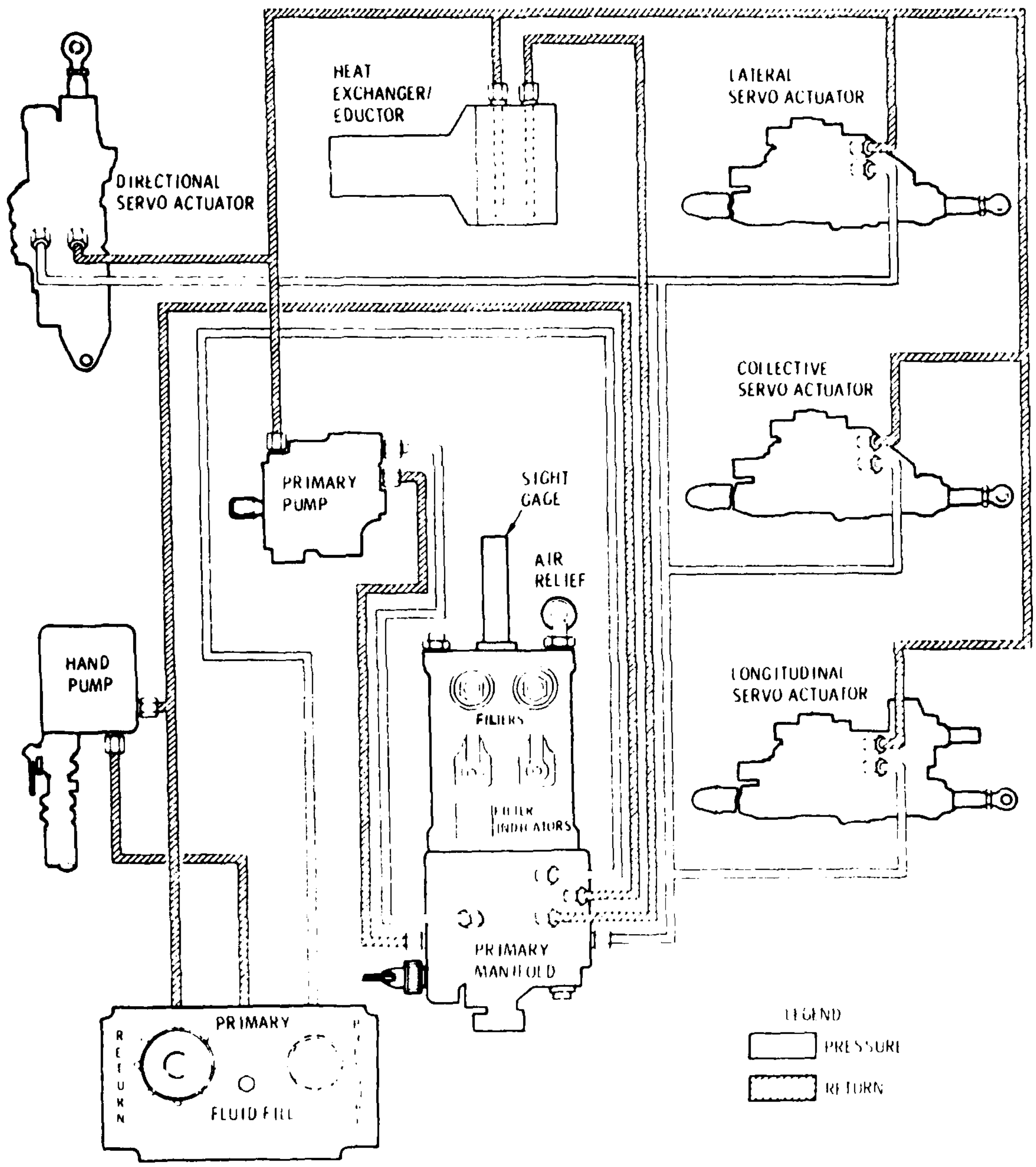


Figure 12. Primary Hydraulic System

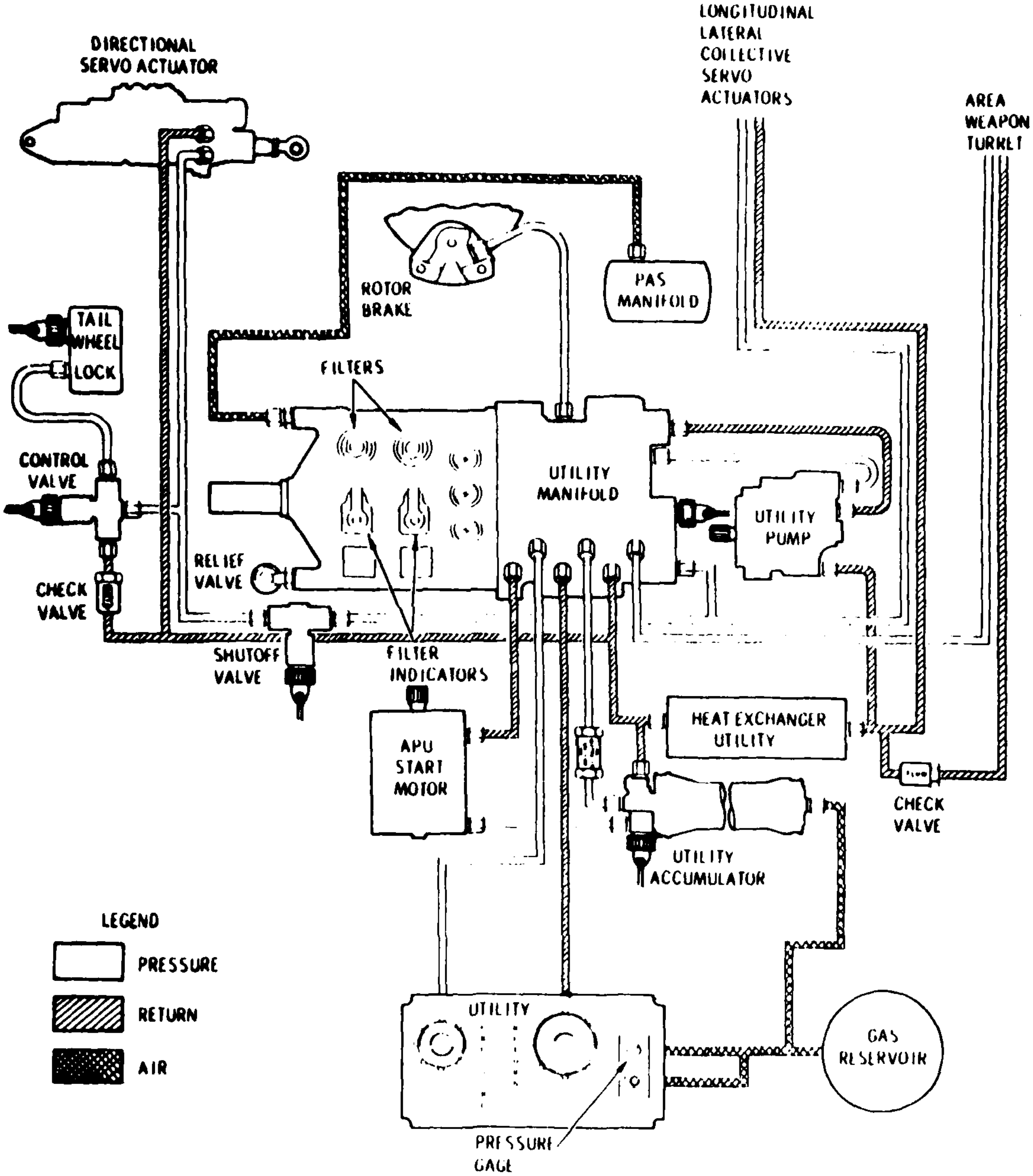


Figure 13. Utility Hydraulic System

Servoactuators

23. Individual hydraulic servoactuators are provided for longitudinal, lateral, collective, and directional controls. Each servoactuator (fig. 14) consists of a ballistically tolerant housing, a single actuator rod and dual frangible pistons, a LAP assembly, BUCS plunger, and various parts for routing of both primary and utility hydraulic fluid. The system is designed to accomodate all flight loads with a failure of either system. However, DASE and BUCS functions would be lost with failure of the primary system. The BUCS plunger assemblies were installed during this test, however, electrical connections were omitted.

POWER PLANT

24. The power plant for the YAH-64 helicopter is the General Electric YT700-GE-700R front drive turboshaft engine, rated at 1563 shp (sea level, standard day, uninstalled). The engines are mounted in nacelles on either side of the main transmission. The basic engine consists of four modules: A cold section, a hot section, a power turbine, and an accessory section. 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. In order to reduce sand and dust erosion, and foreign object damage, an integral particle separator operates when the engine is running. The YT700-GE-700R engine also incorporates a history recorder which records total engine events. Engines S/N 207-239R and 207-258R were installed in the left and right positions, respectively. Pertinent engine data are shown below:

Model	YT700-GE-700R
Type	Turboshaft
Rated power (intermediate)	1563 shp sea level, standard day, uninstalled
Output speed (at 100 percent N_R)	20,952 RPM
Compressor	5 axial stages, 1 centrifugal stage
Variable geometry	Inlet guide vanes, stages 1 and 2 Stator vanes
Combustion chamber	Single annular chamber with axial flow
Gas generator turbine stages	2
Power turbine stages	2
Direction of rotation (aft looking forward)	Clockwise

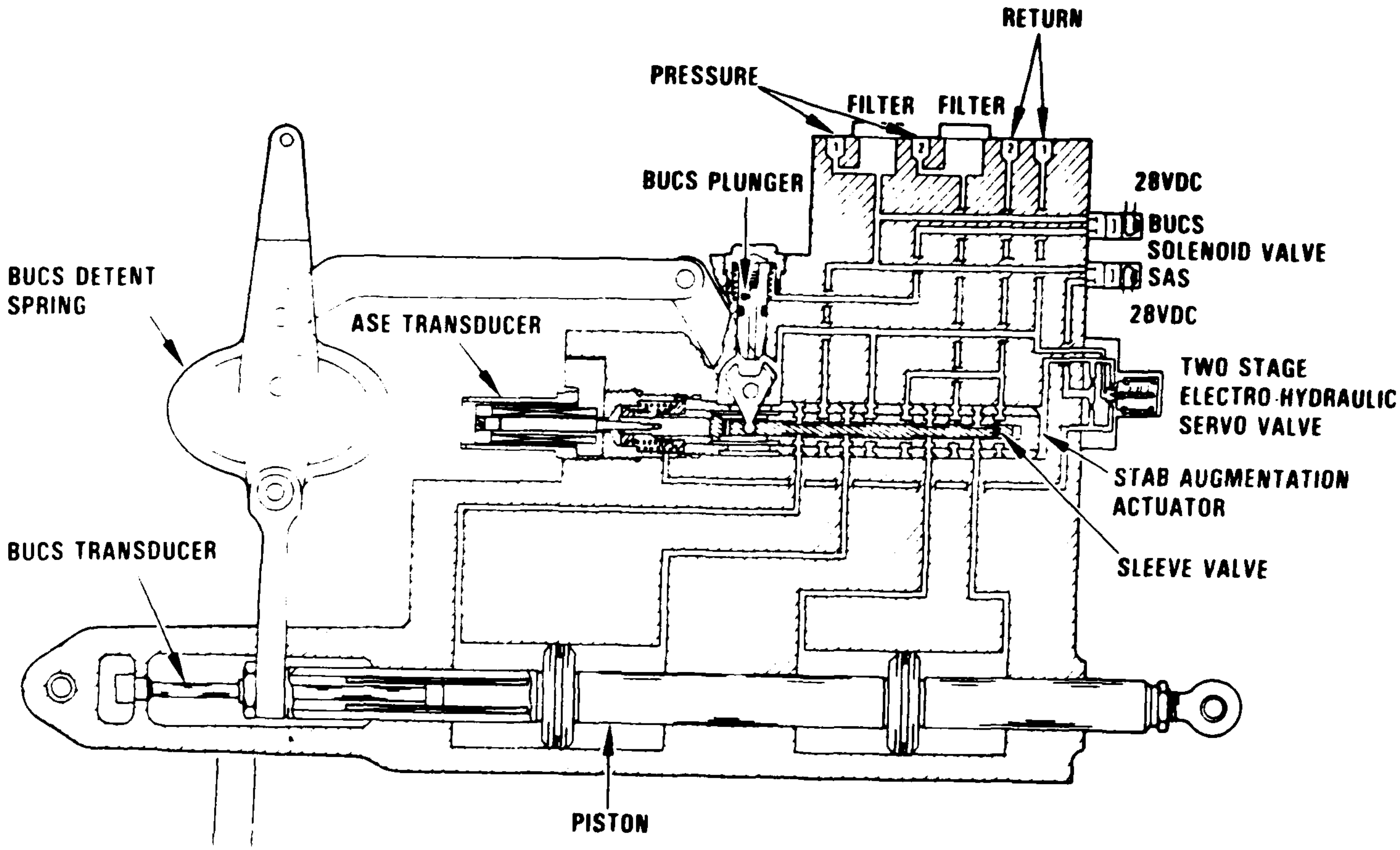


Figure 14. Flight Control Servoactuator

Weight (dry)	415 lb
Length	46.5 in.
Maximum diameter	25 in.
Fuel	MIL-T-5624 (JP-4 or JP-5)
Lubricating oil	MIL-L-7808 or MIL-L-23699
Electrical power requirements for history recorder and N_p overspeed protection	40W, 115 VAC, 400 Hz
Electrical power requirements for anti-ice valve, filter bypass indication, oil filter bypass indication, and magnetic chip detector	1 amp, 28 VDC

INFRARED (IR) SUPPRESSION SYSTEM

25. The IR suppression consists of finned exhaust pipes attached to the engine outlet and bent outboard to mask hot engine parts. The finned pipes radiate heat which is cooled by rotor downwash in hover and turbulent air flow in forward flight . The engine exhaust plume is cooled by mixing it with engine cooling air and bay cooling air (fig. 15). The exhaust acts as an eductor, creating air flow over the combustion section of the engine providing engine cooling. Fixed louvers on the top and bottom of the aft cowl and a door on the bottom forward cowling provide convective cooling to the engine during shutdown. The movable bottom door is closed by engine bleed air during engine operation.

FUEL SYSTEM

26. The YAH-64 fuel system has two fuel cells located fore and aft of the ammunition bay. The system includes a fuel boost pump in the aft cell for starting and for high-altitude operation, a fuel transfer pump for transferring fuel between cells, a fuel crossfeed/shutoff valve, and provisions for pressure and gravity fueling and defueling. Additionally, provisions exist for external, wing-mounted fuel tanks. Figure 16 is a schematic of the fuel system. Figure 17 shows the locations and capacities of the two internal fuel cells.

27. By using the tank select switch on his fuel control panel (fig. 18), the pilot can select either or both tanks from which the engines will draw fuel. With the tank select switch in the NRML position, the left (No. 1) engine will draw fuel from the forward fuel cell and the right (No. 2) engine will draw from the aft cell (fig. 19). When FROM FWD is selected on the tank select switch, the two fuel crossfeed/shutoff valves are positioned so that both engines draw fuel from the forward tank (fig. 20). The FROM AFT position allows the engines to draw fuel from the aft tank only (fig. 21). The tank select switch is disabled whenever the boost pump is on. When the boost pump is on, the fuel crossfeed/shutoff valves are positioned to allow only fuel from the aft cell to feed both engines (fig. 22). The air-driven boost pump operates automatically during engine start and may be activated by the switch on the pilot or CPG fuel control panel (fig. 18).

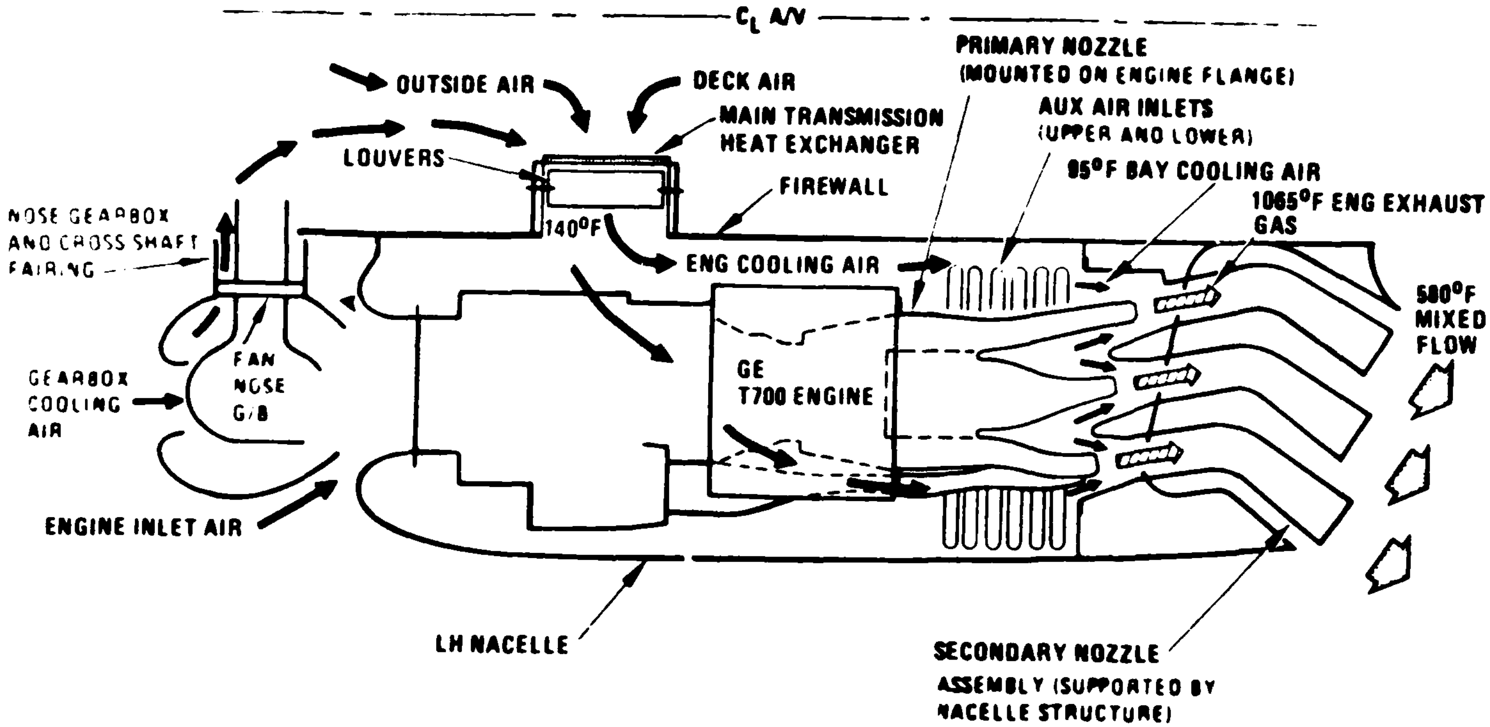


Figure 15. Infrared Suppression System Engine Cooling

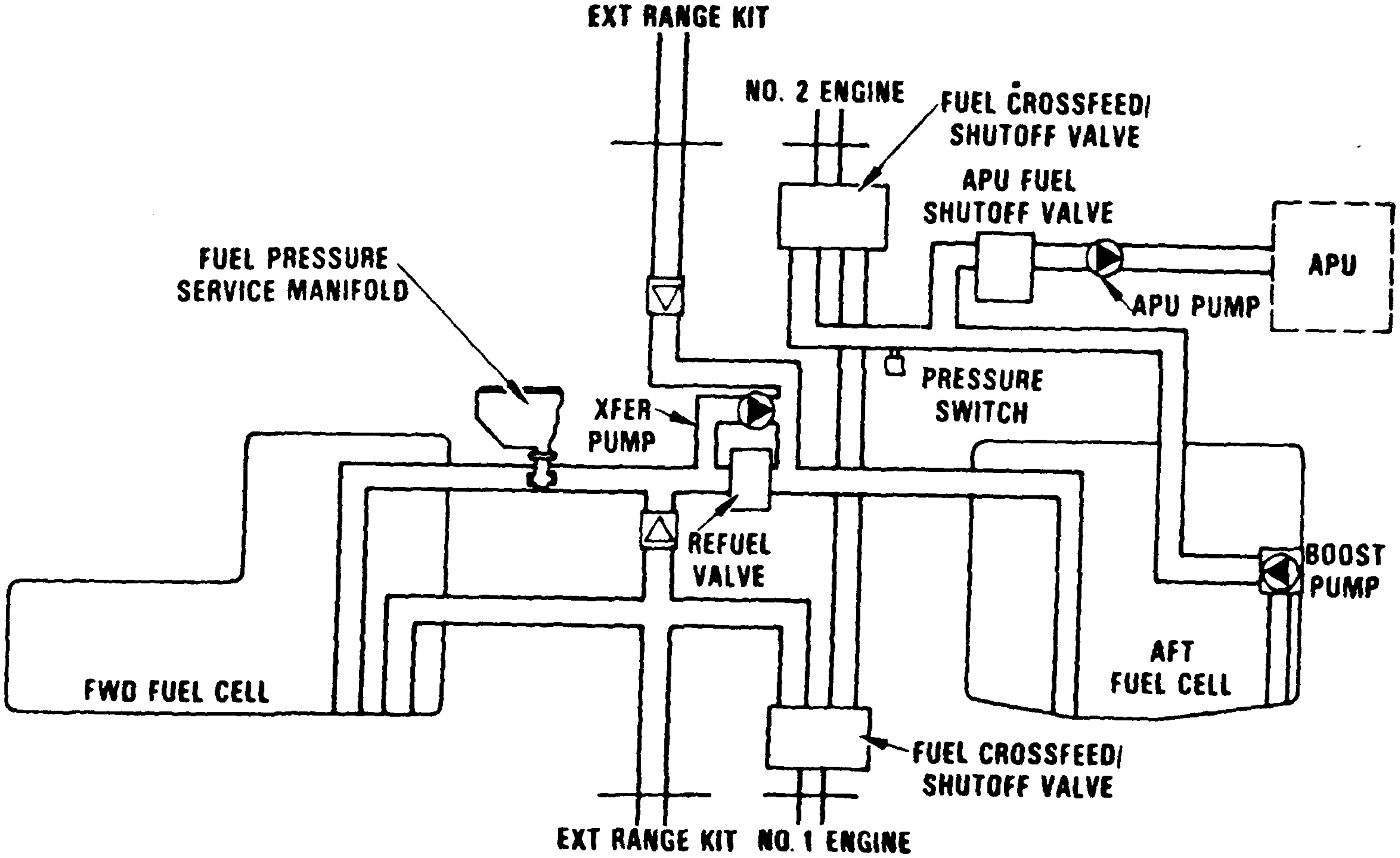


Figure 16. Fuel System Major Components

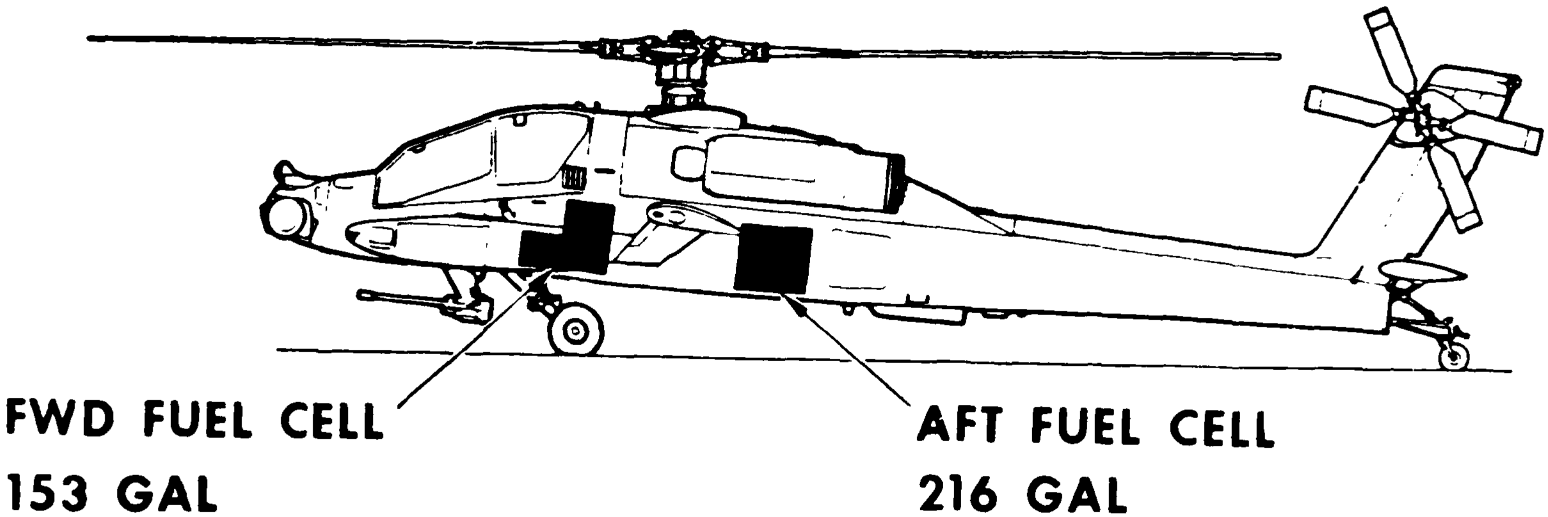


Figure 17. Fuel Cell Locations

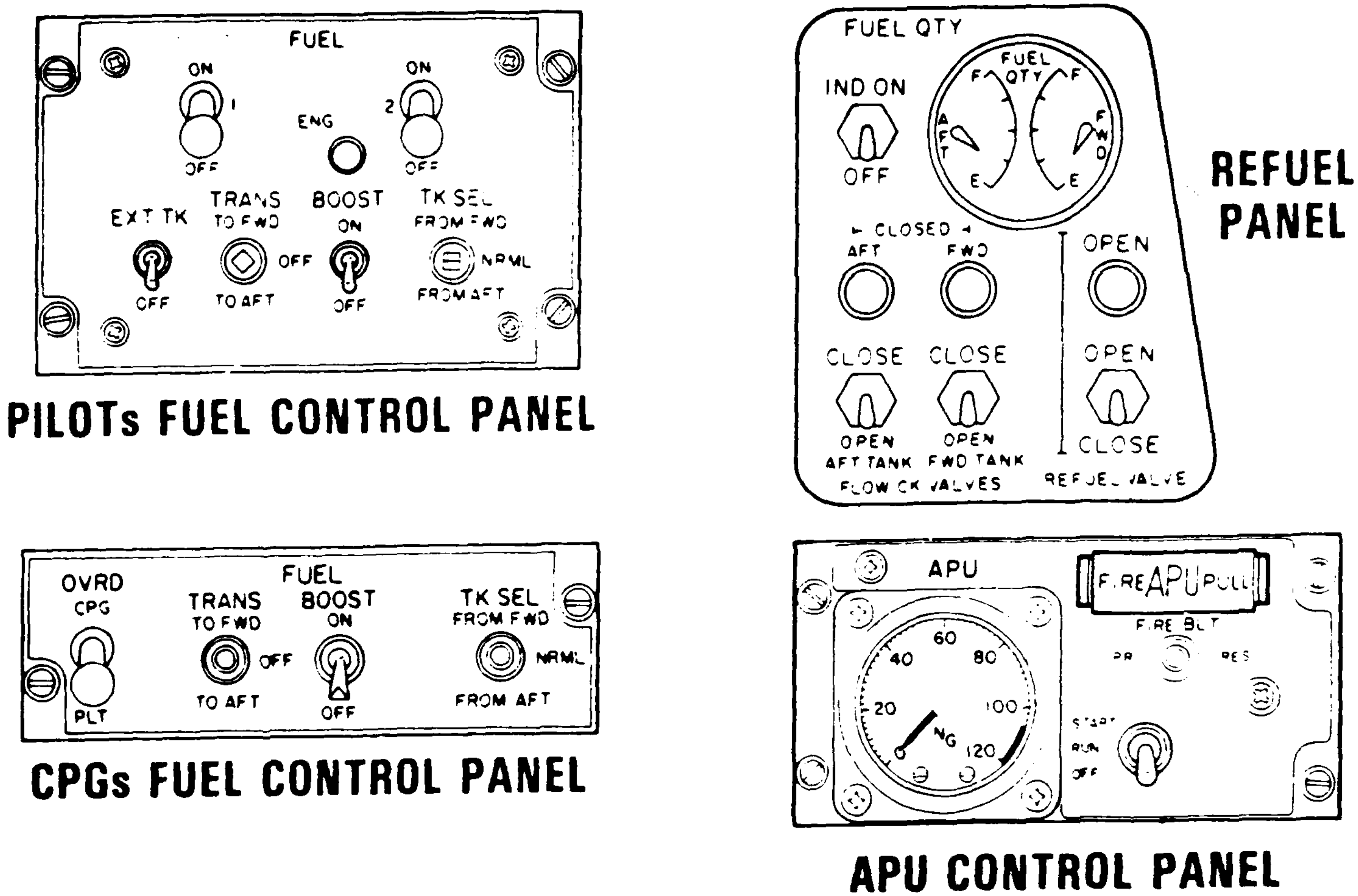


Figure 18. Fuel System Controls

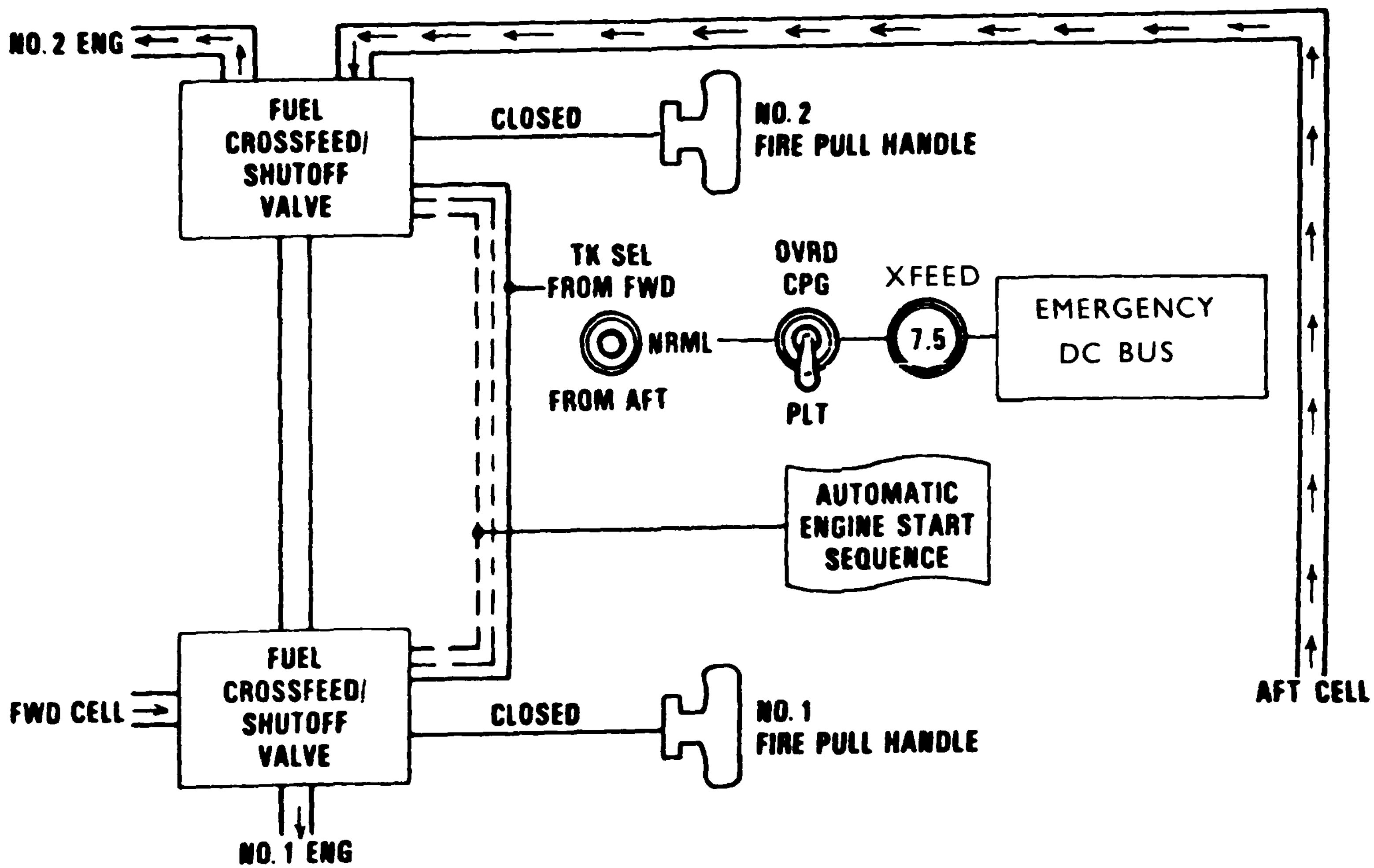


Figure 19. Fuel Crossfeed Operation Normal

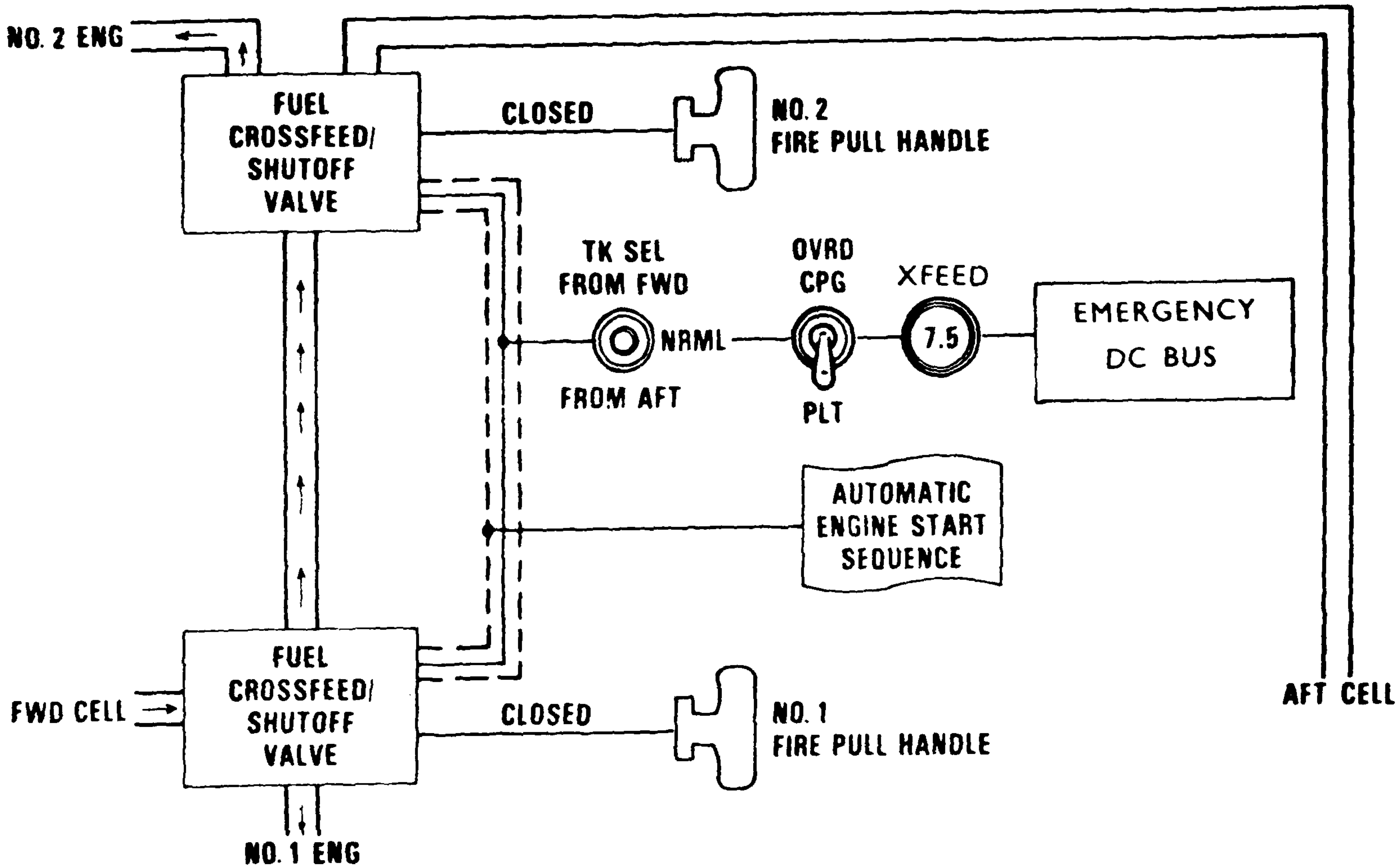


Figure 20. Fuel Crossfeed Operation from Fwd

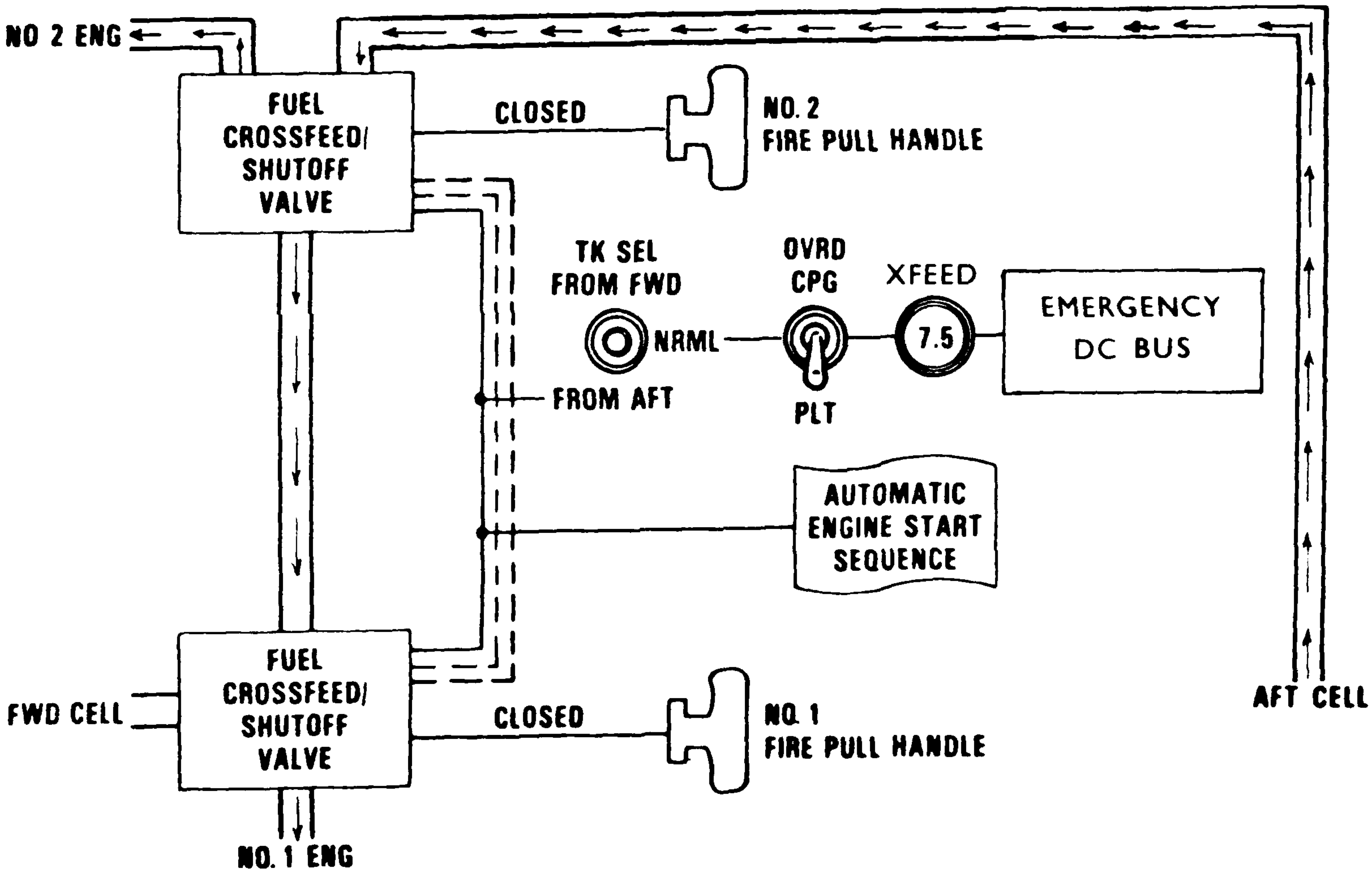


Figure 21. Fuel Crossfeed Operation from Aft

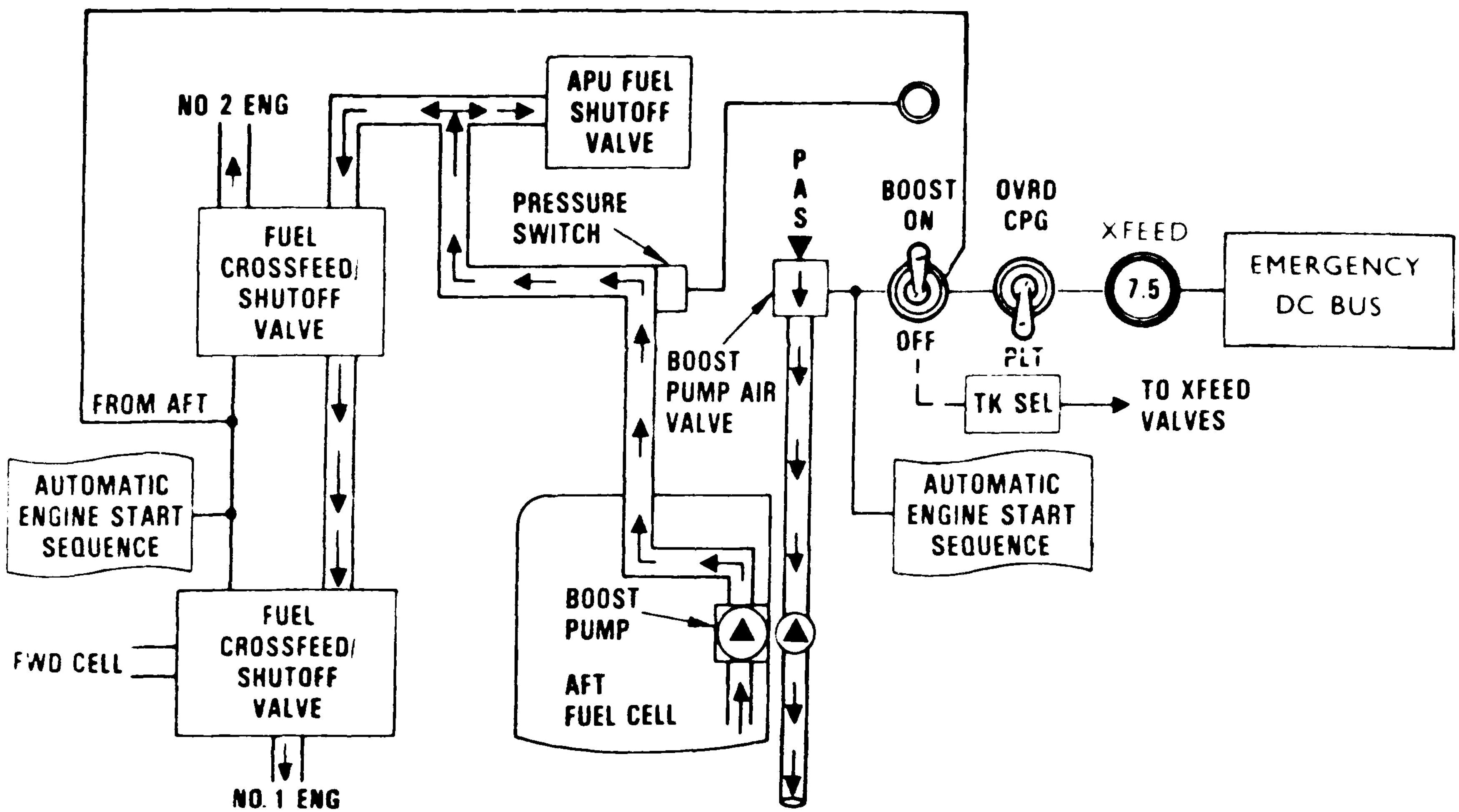


Figure 22. Fuel System Boost Operation

28. The pilot and CPG also have the capability to transfer fuel between tanks using the transfer switch on the fuel control panels (fig. 18). Moving the fuel transfer switch out of the OFF position closes the refuel valve and starts the air-driven pump transferring fuel in the selected direction (fig. 23).

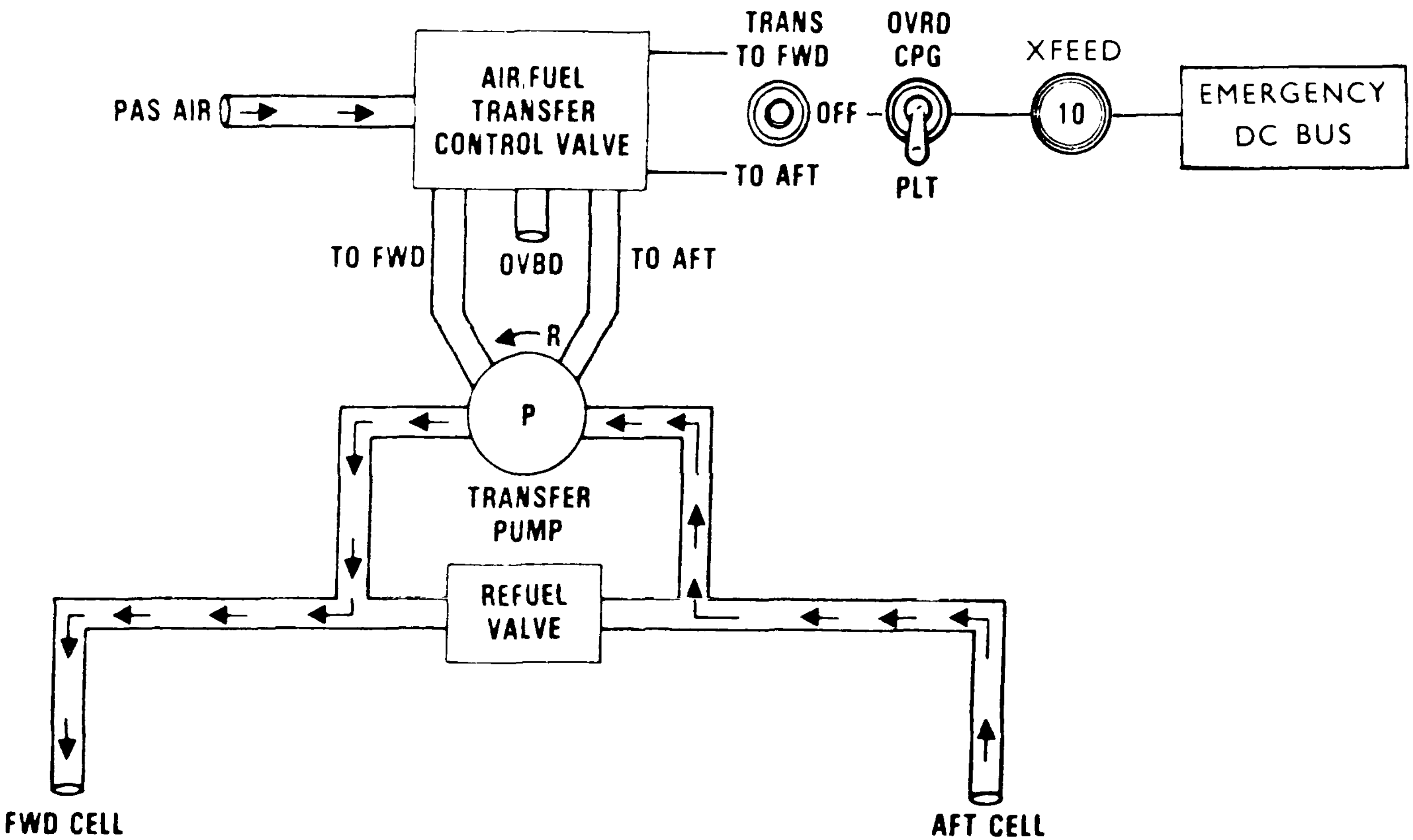


Figure 23. Fuel Transfer Operation

APPENDIX C. INSTRUMENTATION

The airborne data acquisition system was installed, calibrated, and maintained by Hughes Helicopters. The system used pulse code modulation (PCM) encoding, and magnetic tape was used to record parameters on board the aircraft. A boom was mounted on the left side of the aircraft, extending 52 inches forward of the nose. A pitot-static tube, an angle-of-attack sensor, and an angle-of-sideslip sensor were mounted on the boom. Calibration of the boom airspeed system is presented in figure 1. Instrumentation and related special equipment used for this test follows:

Pilot Station (Aft Cockpit)

Pressure altitude (ship)
Airspeed (boom) (sensitive and digital)
Tether cable tension
Main rotor speed (digital)
Engine torque (both engines)* Engine turbine gas temperature (both engines)*
Engine power turbine speed (both engines)*
Engine gas producer speed (both engines)*
Angle of sideslip
Event switch
Tether cable angles (longitudinal and lateral)
Longitudinal control position
Lateral control position
Directional control position
Collective control position
Stabilator angle*
Normal acceleration

Copilot/Engineer Station

Airspeed (left pitot)
Altitude (ship)
Main rotor speed
Engine torque (both engines)*
Engine turbine gas temperature (both engines)*
Engine gas producer speed (both engines)*
Total air temperature
Time code display
Event switch
Data system controls
Fuel used (both engines)

PCM Parameters

Time code
Event
Main rotor speed
Fuel temperature (both engines)
Fuel used (both engines)
Engine fuel flow rate (both engines)
Engine gas producer speed (both engines)
Engine power turbine speed (both engines)
Engine torque (both engines)

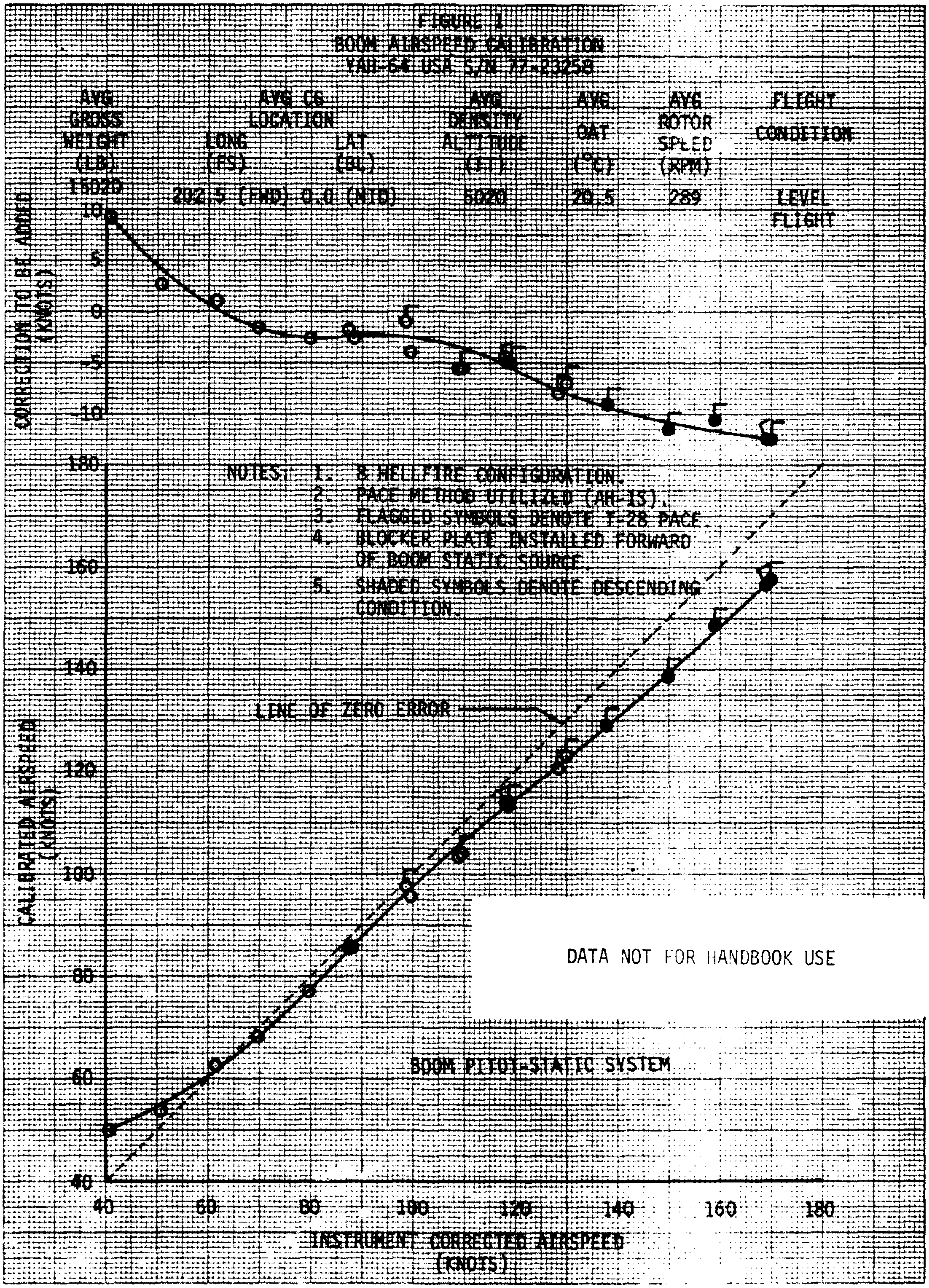


FIGURE 1
BOOM AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258

Engine turbine gas temperature (both engines)
Airspeed (boom)
Airspeed (ship, right and left)
Altitude (boom)
Altitude (ship)
Total air temperature
Angle of attack
Angle of sideslip
Tether cable tension
Tether cable angle (longitudinal and lateral)
Control positions:
- Longitudinal cyclic
- Lateral cyclic
- Directional
- Collective
- Stabilator

Aircraft attitudes:
- Pitch
- Roll
- Yaw

Aircraft angular velocities:
- Pitch
- Roll
- Yaw

Stability augmentation system actuator positions:
- Longitudinal
- Lateral
- Directional

Air data system:
- Longitudinal airspeed
- Lateral airspeed
- Resultant airspeed
- Pressure altitude
- Air Temperature

Control actuator positions:
- Tail rotor
- Collective pitch
- Longitudinal cyclic
- Lateral cyclic

Main rotor shaft torque
Tail rotor shaft torque
Radar altitude
Center of gravity normal acceleration
Vibration accelerometers:
- Pilot station (3 axes)(pilot seat)
- Pilot floor (3 axes)
- Copilot station (3 axes)(copilot seat)
- Center of gravity (3 axes)

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. The helicopter performance test data were generalized by use of nondimensional coefficients and were such that the effects of compressibility and blade stall were not separated and defined. The following nondimensional coefficients were used to generalize the hover and level flight test results obtained during this flight test program.

a. Coefficient of power (C_P):

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

b. Coefficient of thrust (C_T):

$$C_T = \frac{\text{Thrust}}{\rho A(\Omega R)^2} \tag{2}$$

c. Advance ratio (μ):

$$\mu = \frac{1.6878\ V_T}{\Omega R} \tag{3}$$

d. Advancing tip Mach number (M_{tip}):

$$M_{tip} = \frac{1.6878\ V_T + (\Omega R)}{a} \tag{4}$$

Where:

SHP = Engine output shaft horsepower (both engines)
550 = Conversion factor (ft-lb/sec)/SHP
ρ = Air density (slug/ft^3)
A = Main rotor disc area (ft^2) = (1809.56)
Ω = Main rotor angular velocity (radian/sec) = 30.26 (at 289 RPM)
R = Main rotor radius (ft) = 24
Thrust = Gross weight (lb) during free flight in which there is no acceleration or velocity component in the vertical direction. Tether load must be added in the case of tethered hover.
1.6878 = Conversion factor (ft/sec)/kt
V_T = True airspeed (kt)
a = Speed of sound (ft/sec) = $1116.45\sqrt{\theta}$
θ = (T + 273.15)/288.15
T = Ambient air temperature

For a rotor speed of 289 RPM, the following constants were used:

$A = 1809.56\ ft^2$
$\Omega R = 726.34\ ft/sec$
$A(\Omega R)^2 = 954657879\ ft^4/sec^2$
$A(\Omega R)^3 = 6.934025959 \times 10^{11}\ ft^5/sec^3$

SHAFT HORSEPOWER REQUIRED

2. Engine output shaft torque was determined by the use of the engine torquemeter. The torquemeter was calibrated in a test cell by the engine manufacturer. The outputs from the engine torquemeters were recorded on the on board data recording system. The output shp was determined from the engine output shaft torque and rotational speed by the following equation:

$$SHP = \frac{2\pi \times N_P \times Q}{33{,}000} \qquad (5)$$

Where:

N_P = Engine output shaft rotational speed (RPM)
Q = Calibrated engine output shaft torque (ft-lb)
33,000 = Conversion factor (ft-lb/min)/shp

HOVER PERFORMANCE

3. Hover performance data were gathered during 5-foot and 100-foot tethered hovering flight. Power was varied between data points from the minimum required to maintain tension in the tether cable, to the maximum power available. Cable tension was measured and added to the aircraft gross weight to determine thrust (required in equation 2). To further increase the range of C_T and C_P, main rotor speed was varied from approximately 97 to 100 percent.

LEVEL FLIGHT PERFORMANCE

4. Level flight performance data were reduced to nondimensional form using equations 1, 2, and 3. Each speed-power was flown at a predetermined C_T with rotor speed held constant. To maintain the ratio of gross weight to air density ratio (W/σ) constant, altitude was increased as fuel was consumed.

5. Test-day (measured) level flight power was corrected to standard-day conditions (average for the flight) by assuming that the test-day dimensionless parameters C_{P_t}, C_{T_t}, and μ_t are identical to C_{P_s}, C_{T_s}, and μ_s, respectively.

From equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \left(\frac{\rho_s}{\rho_t}\right) \quad (6)$$

Where:

subscript t = Test day
subscript s = Standard day

6. Test specific range was calculated using level flight performance curves and the measuerd fuel flow.

$$SR = \frac{V_T}{W_f} \quad (7)$$

Where:

SR = Specific range (nautical air miles per pound of fuel)
V_T = True airspeed (kt)
W_f = Fuel flow (lb/hr)

HANDLING QUALITIES

7. Stability and control data were collected and evaluated using standard test methods as described in reference 11, appendix A. Definitions of deficiencies and shortcomings used during this test are shown below.

Deficiency - A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued; or indicates improper design or other cause of failure of an item or part, which seriously impairs the equipment's operational capability.

Shortcoming - An imperfection or malfunction occurring durnng the life cycle of equipment which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the useability of the material or end product.

VIBRATIONS

8. The PCM vibration data were reduced by means of a fast Fourier transform from the analog flight tape. Vibration levels, representing peak amplitudes, were extracted from this analysis at selected harmonics of the main rotor frequency. The Vibration Rating Scale, presented in figure 1, was used to augment crew comments on aircraft levels.

DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

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

Figure 1. Vibration Rating Scale

AIRSPEED CALIBRATION

9. The boom pitot-static system and both ship's systems were calibrated by using the pace aircraft method to determine the airspeed position error. Calibrated airspeed (V_{cal}) was obtained by correcting indicated airpseed (V_i) using instrument (ΔV_{ic}) and position (ΔV_{pc}) error corrections.

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc} \tag{8}$$

10. True airspeed (V_t) was calculated from the calibrated airpseed and density ratio.

$$V_t = \frac{V_c}{\sqrt{\sigma}} \tag{9}$$

Where:

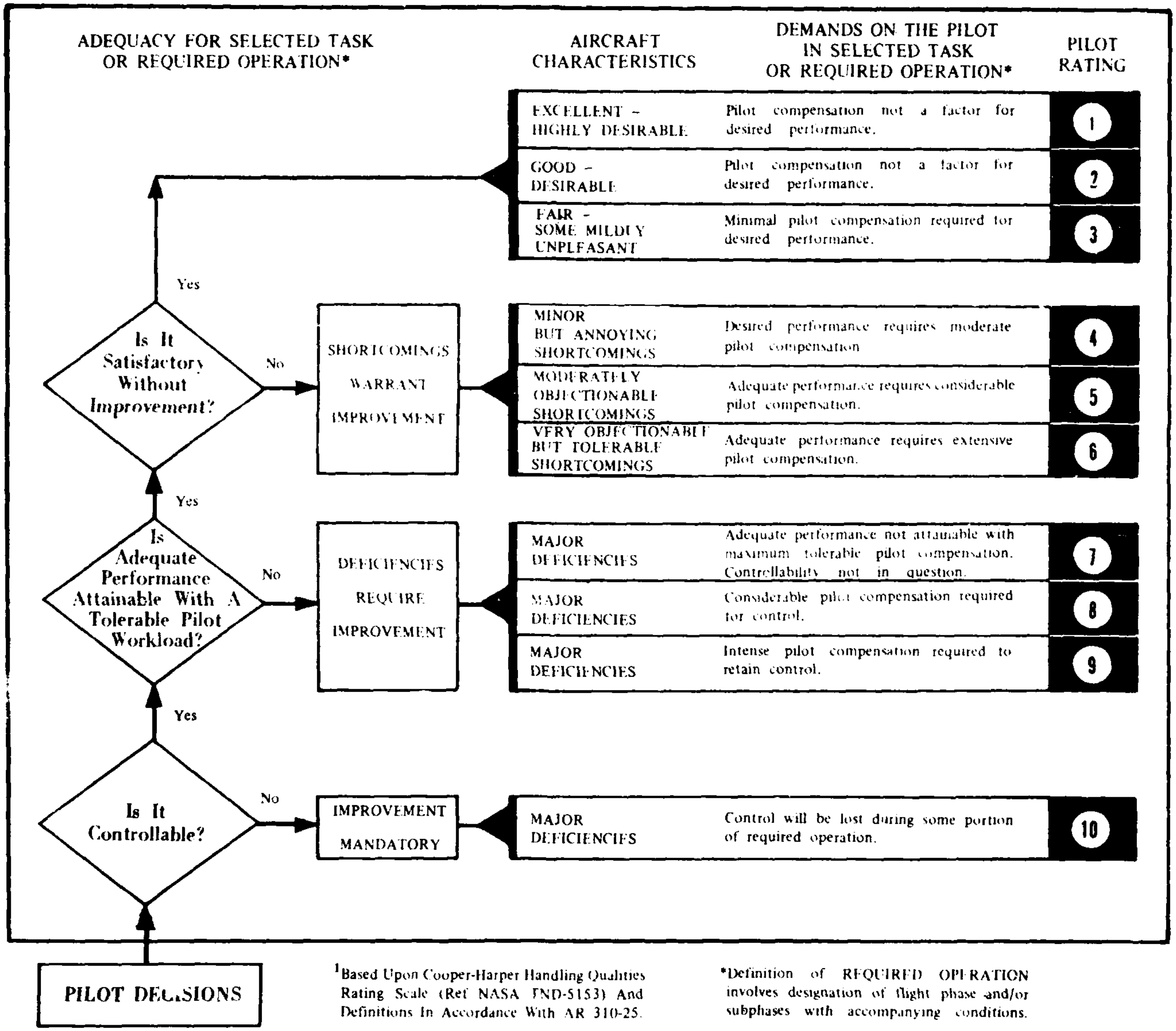
σ = Density ratio ($\frac{\rho}{\rho_o}$) where ρ_o is the density at sea level on a standard day.

WEIGHT AND BALANCE

11. Prior to testing the aircraft gross weight and cg were determined by using calibrated scales. The aircraft was weighed with no fuel, in the clean configuration, with instrumentation on board. The aircraft weight was 11,861 pounds with a longitudinal cg location at FS 214.1.

HANDLING QUALITITES RATING SCALE

12. The Handling Qualities Rating Scale (HQRS) presented in figure 2 was used to augment pilot comments relative to handling qualities and work load.



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure 2. Handling Qualities Rating Scale

APPENDIX E. TEST DATA

INDEX

FIGURE 1
NONDIMENSIONAL HOVER PERFORMANCE
YAH-64 USA S/N 77-23258

WHEEL HEIGHT = 5 FEET

SYMBOL	AVG ROTOR SPEED (RPM)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)
□	289	6320	25.0
△	281	6340	25.5
⬭	290	11440	13.5
▽	282	11440	13.5
◇	280	11440	13.5

NOTES: 1. WINDS 3 KNOTS OR LESS.
2. TETHERED HOVER TECHNIQUE.
3. SHADED SYMBOLS DENOTE FREE FLIGHT TECHNIQUE.
4. CLEAN CONFIGURATION.

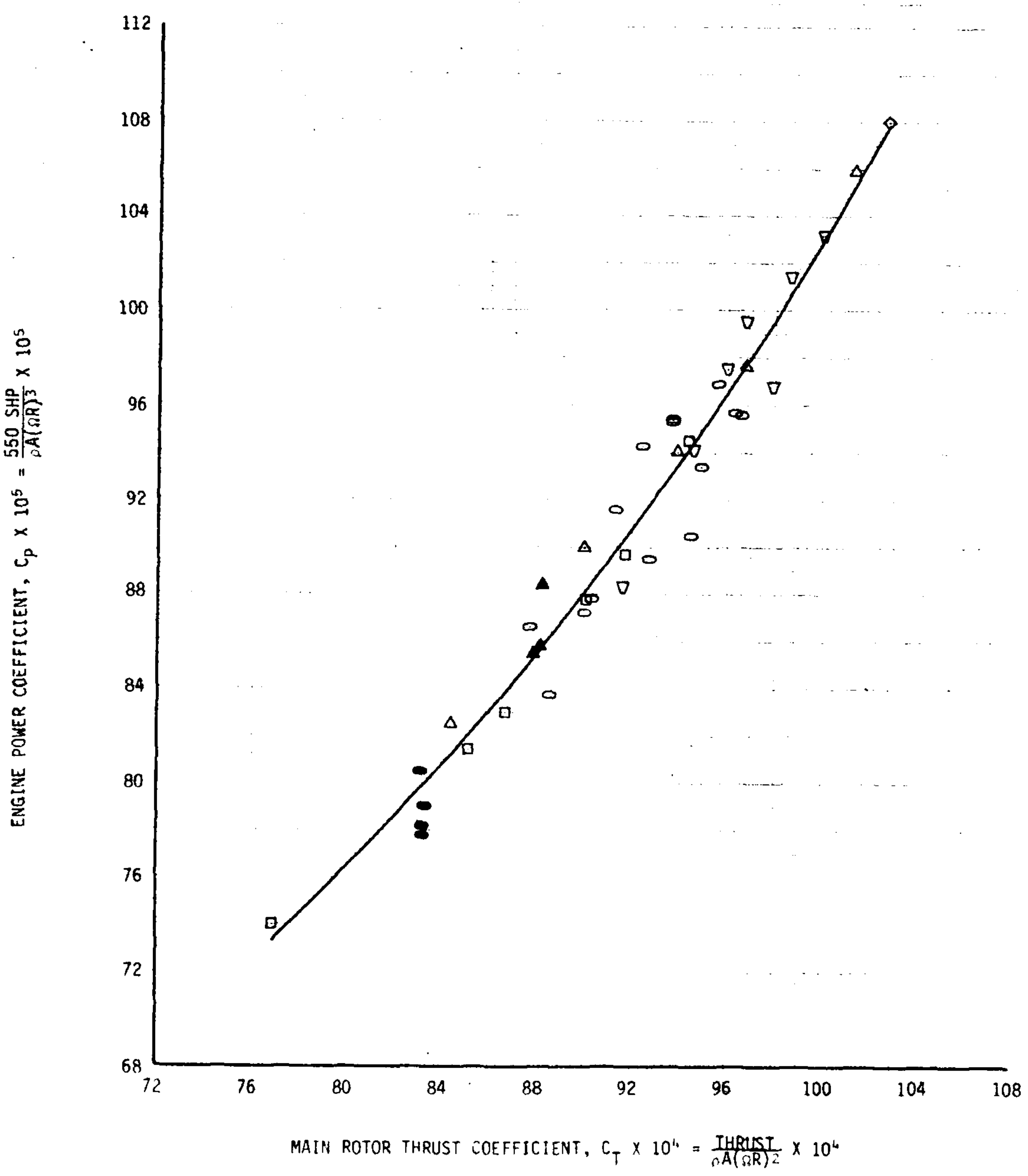


FIGURE 2
NONDIMENSIONAL HOVER PERFORMANCE
YAH-64 USA S/N 77-23258

WHEEL HEIGHT = 100 FEET

SYMBOL	AVG ROTOR SPEED (RPM)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)
□	282	5800	19.0
△	289	5820	19.0
○	289	11600	13.0
▽	282	11700	13.5
◇	285	11800	14.0
◺	286	11860	14.0

NOTES:
1. TETHERED HOVER TECHNIQUE
2. SHADED SYMBOLS DENOTE FREE FLIGHT TECHNIQUE.
3. CLEAN CONFIGURATION.
4. WINDS 3 KNOTS OR LESS.

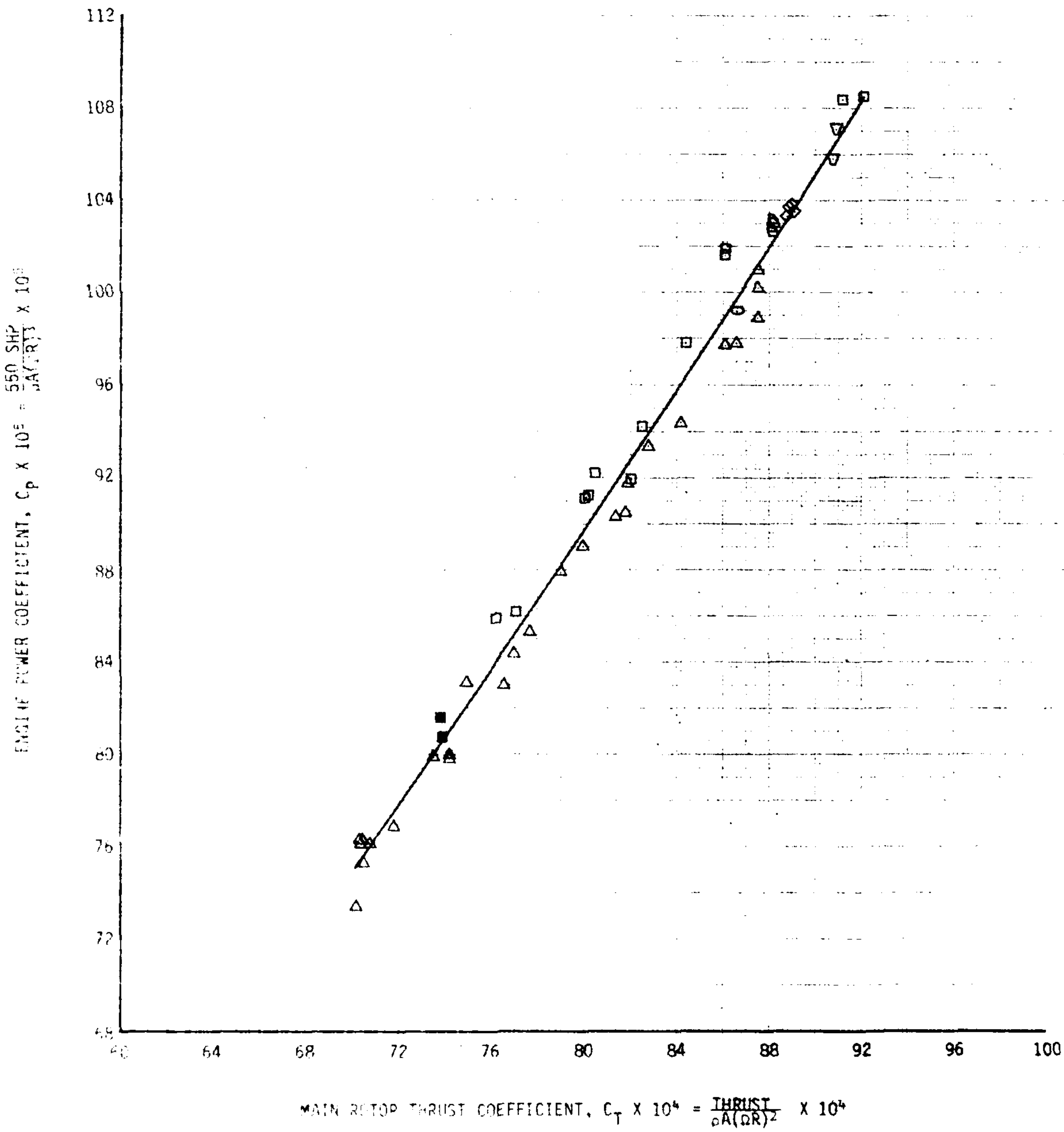


FIGURE 3
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 8 THROUGH 10

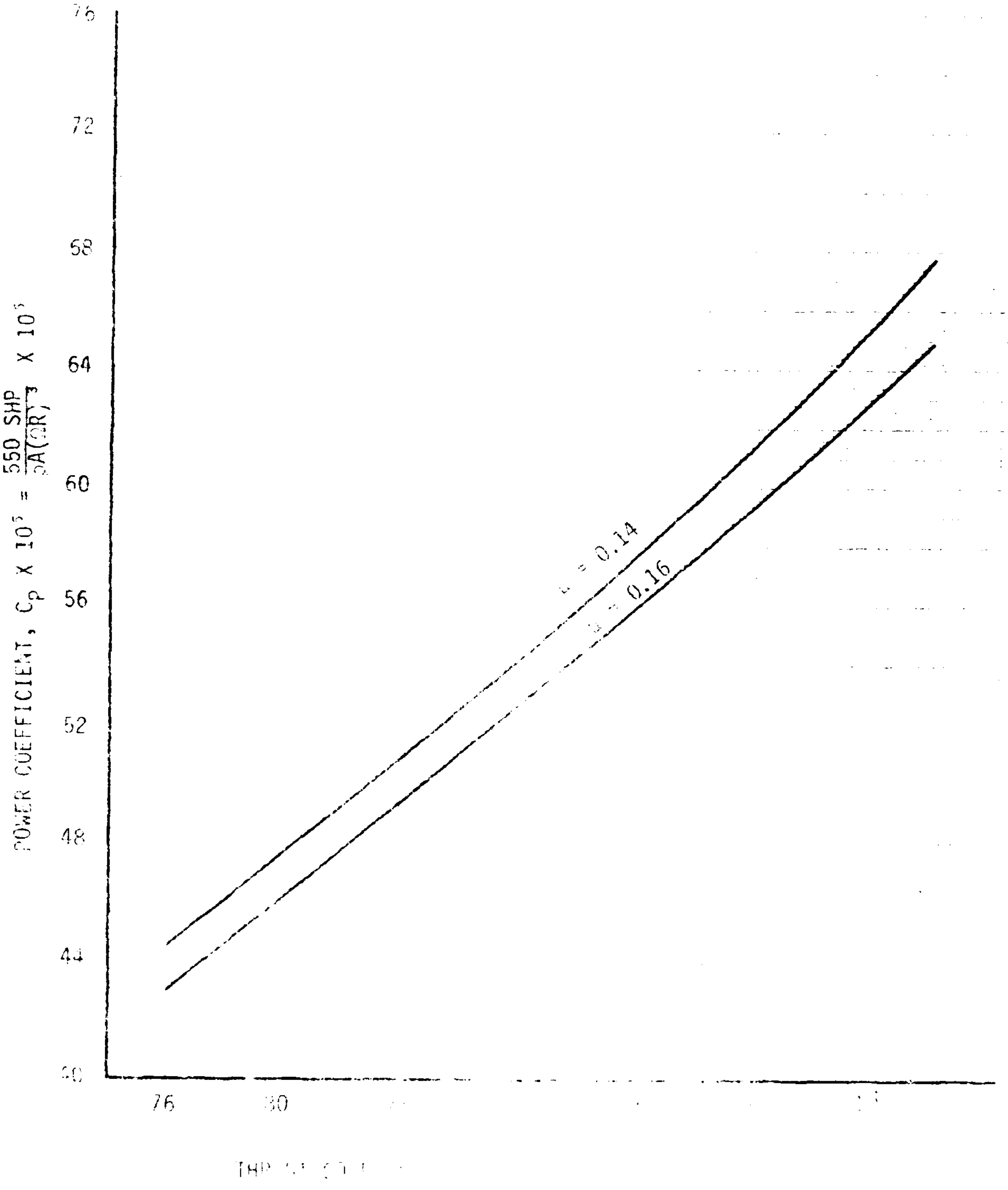


FIGURE 4
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES:
1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 8 THROUGH 10

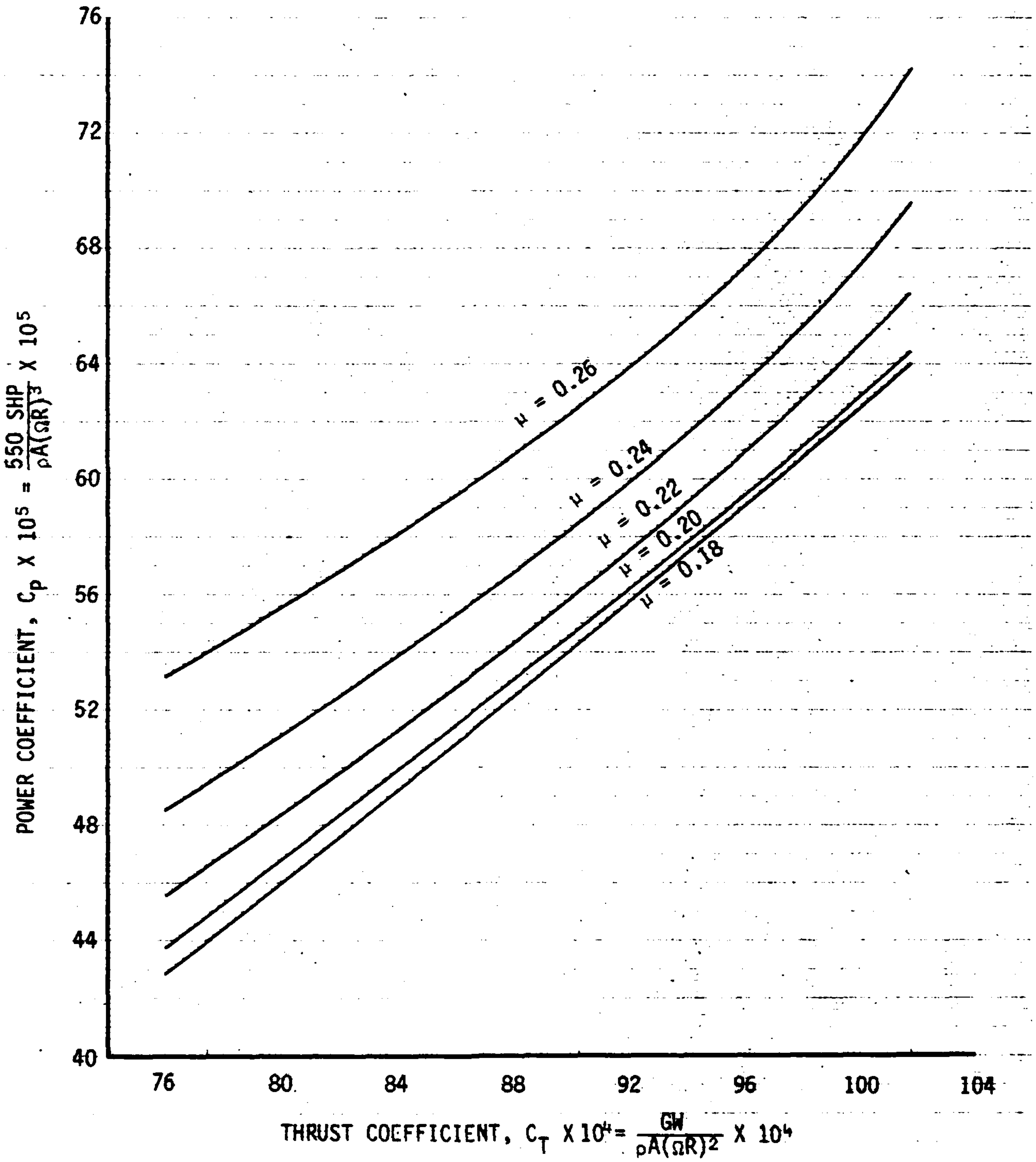


FIGURE 5
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES:
1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 8 THROUGH 10

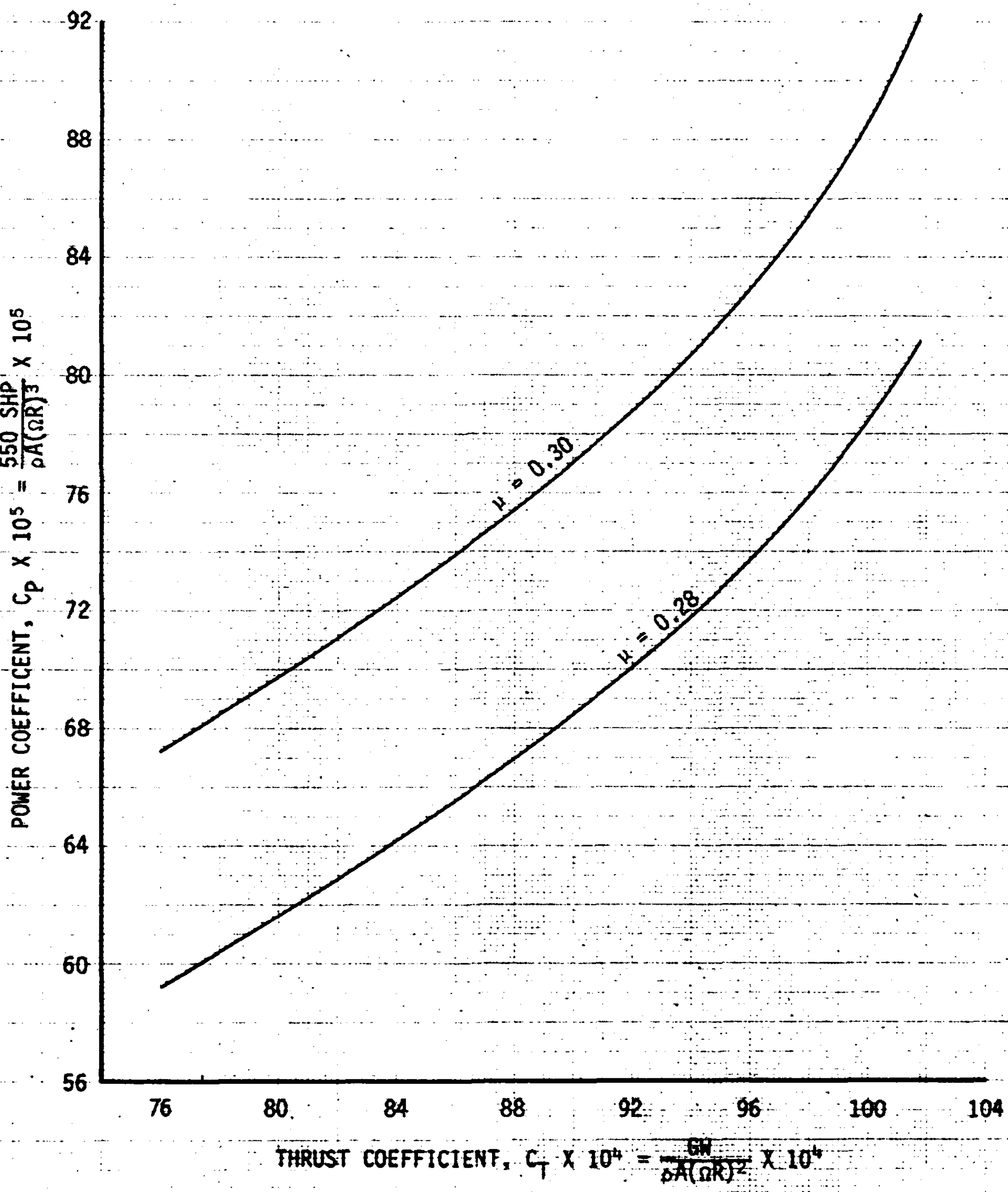


FIGURE 6

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 11 THROUGH 13

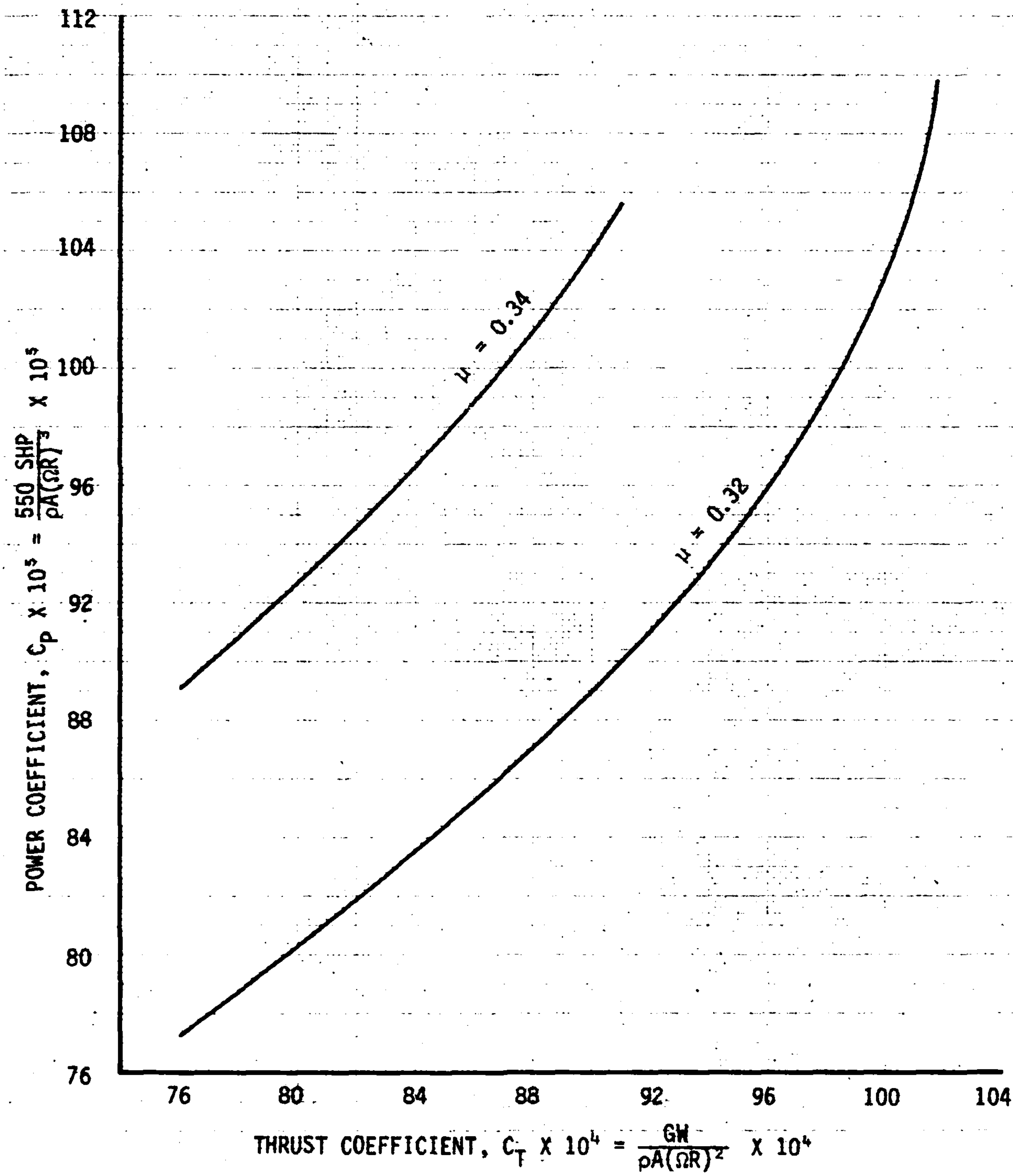


FIGURE 7
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG LONG. CG LOCATION (FS)	AVG PRESSURE ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
14525	202.0 (FWD)	4000	35.0	289	0.007924	8-HELLFIRE

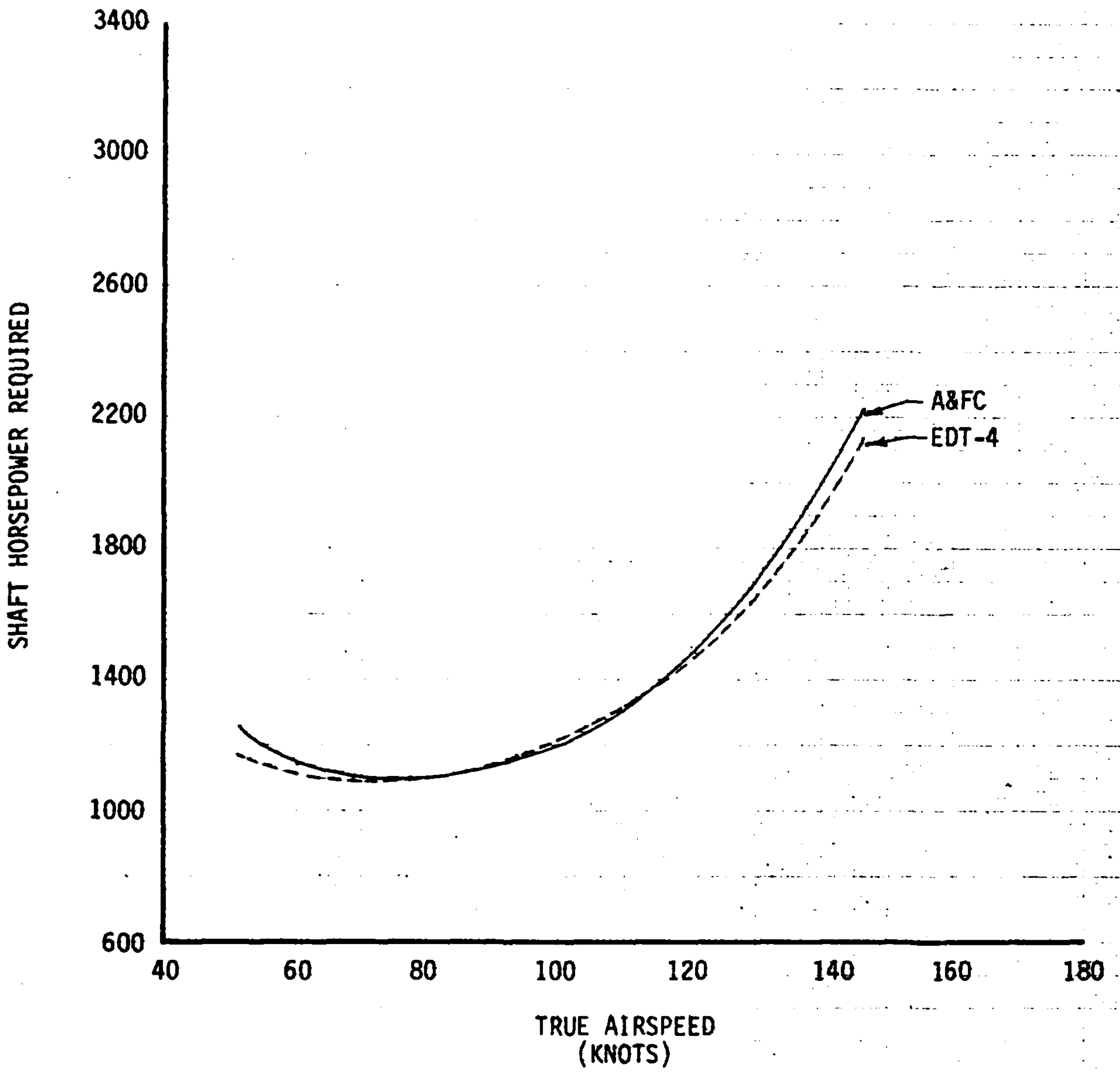


FIGURE 8
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

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 X10⁴	CONFIGURATION
14960	202.0 (FWD)	0.0 (MID)	5620	29.5	290	0.007743	8-HELLFIRE

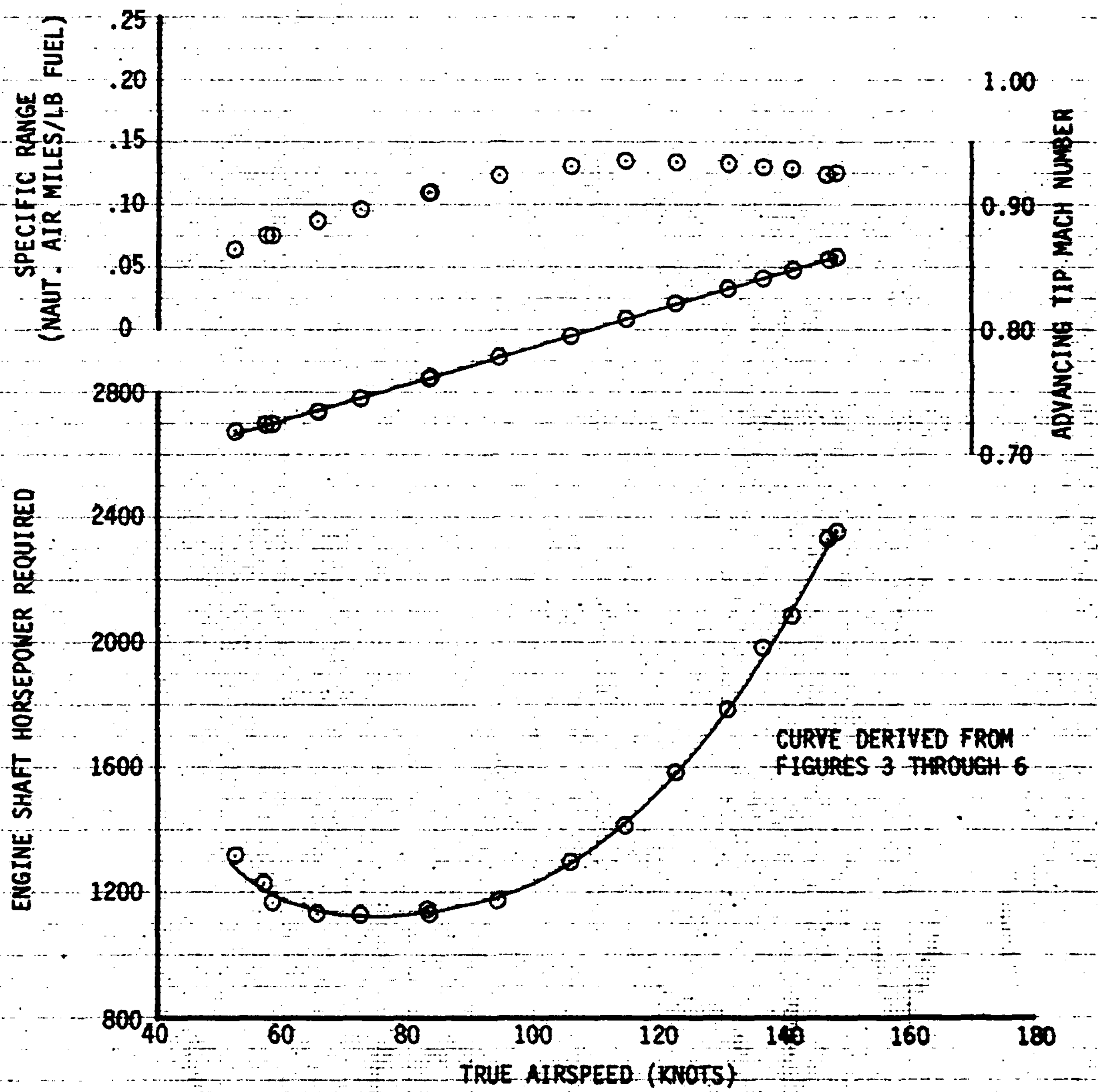


FIGURE 9
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

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	CONFIGURATION
15500	202.2 (FWD)	0.0 (MID)	9460	15.0	290	0.009031	8-HELLFIRE

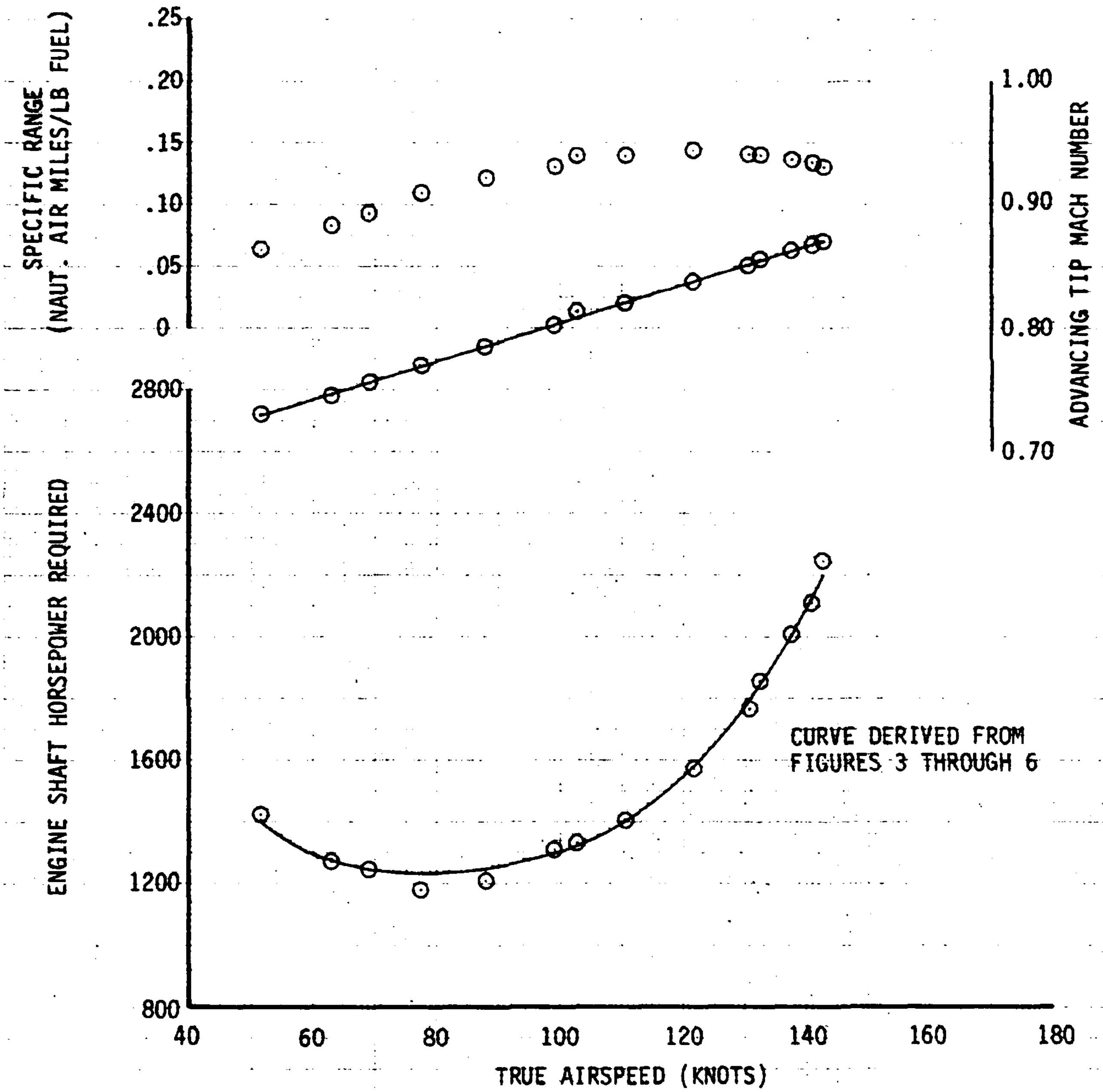


FIGURE 10
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

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 $X10^4$	CONFIGURATION
15400	201.9 (FWD)	0.0 (MID)	13220	5.5	289	0.010178	8-HELLFIRE

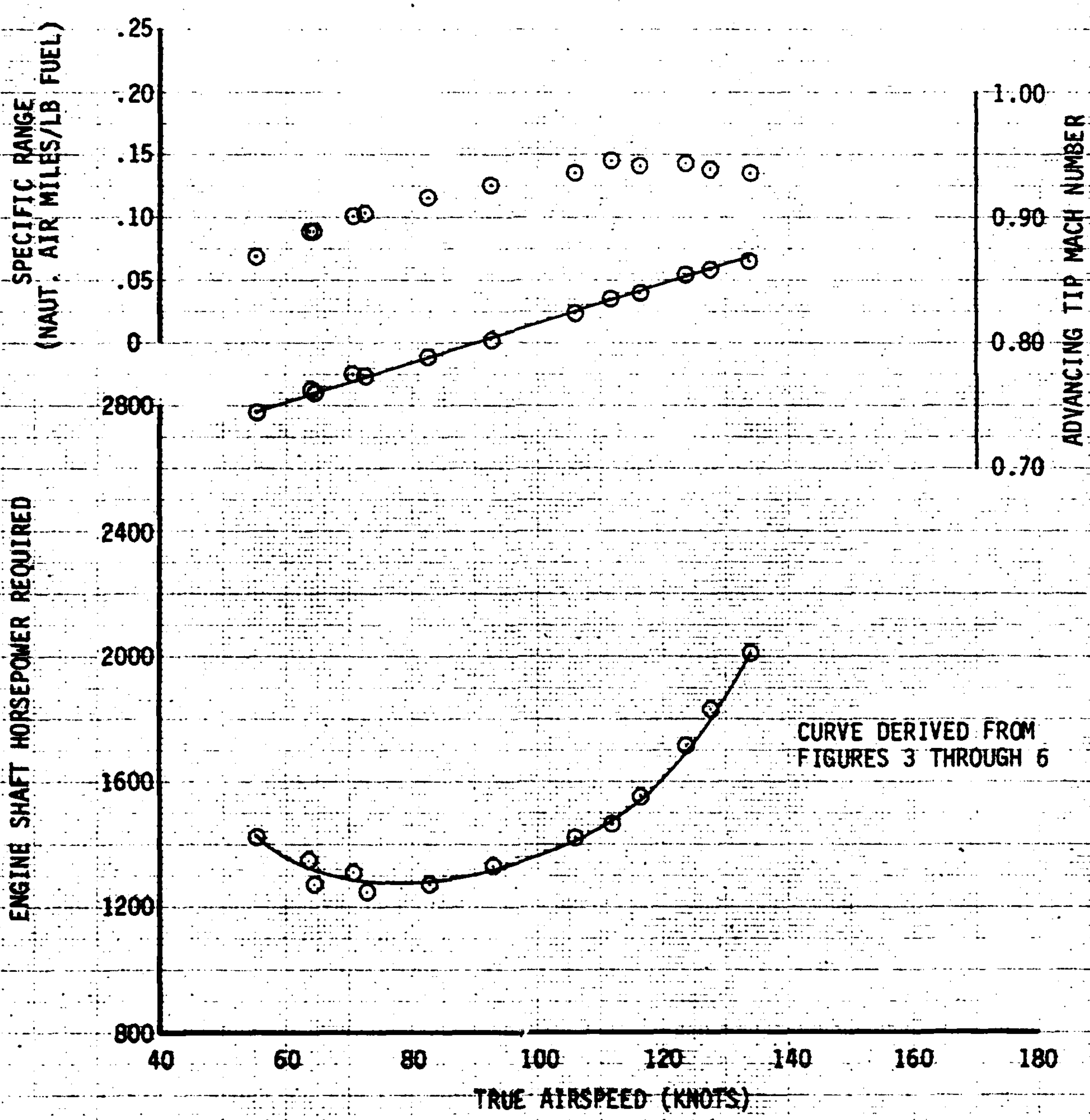


FIGURE 11
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

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 X10⁴	CONFIGURATION
15780	201.3 (FWD)	0.0 (MID)	7300	20.5	290	0.008598	16-HELLFIRE

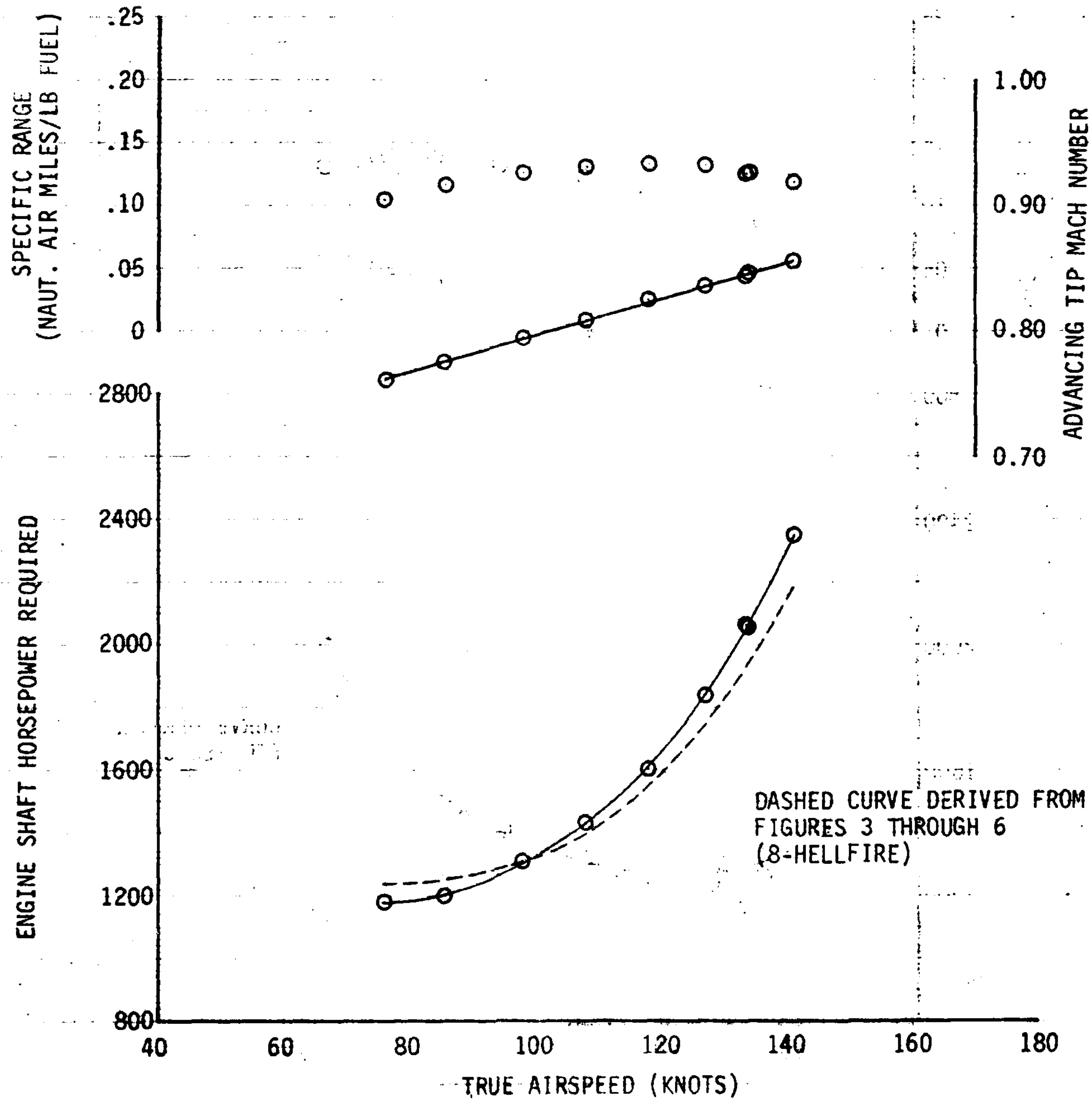


FIGURE 12
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

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 X10⁴	CONFIGURATION
16420	201.4 (FWD)	0.0 (MID)	10440	11.0	290	0.009866	16-HELLFIRE

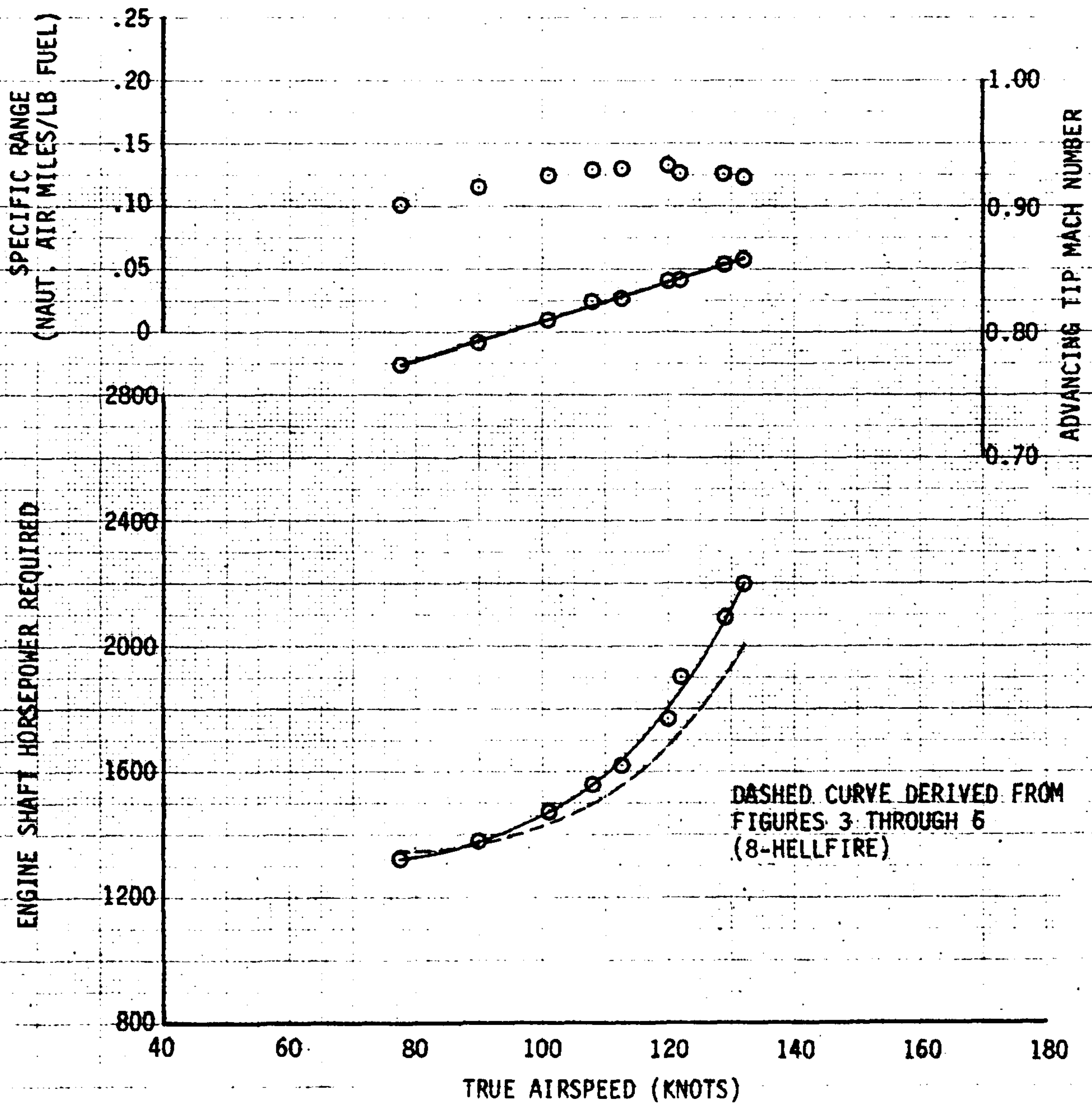


FIGURE 13
CONTROL POSITION IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

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
15020	202.5 (FWD)	0.0(MID)	5020	20.5	289	LEVEL

NOTES: 1. 8-Hellfire configuration.
2. Shaded symbols denote descending condition

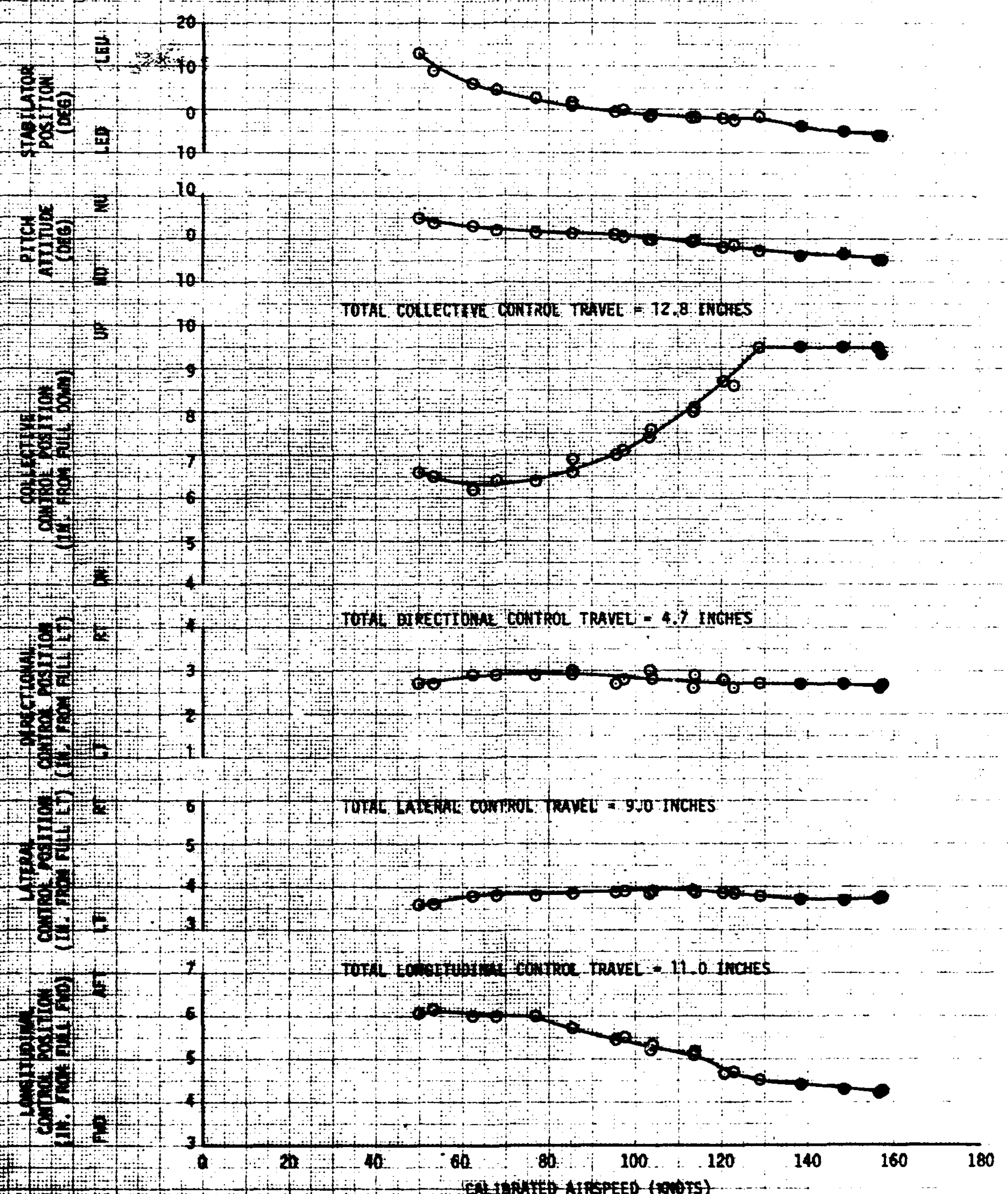


FIGURE 14
CONTROL POSITION IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

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
15760	201.3(FWD)	0.0(MID)	7300	20.5	289	LEVEL

NOTE: 16-HELLFIRE CONFIGURATION

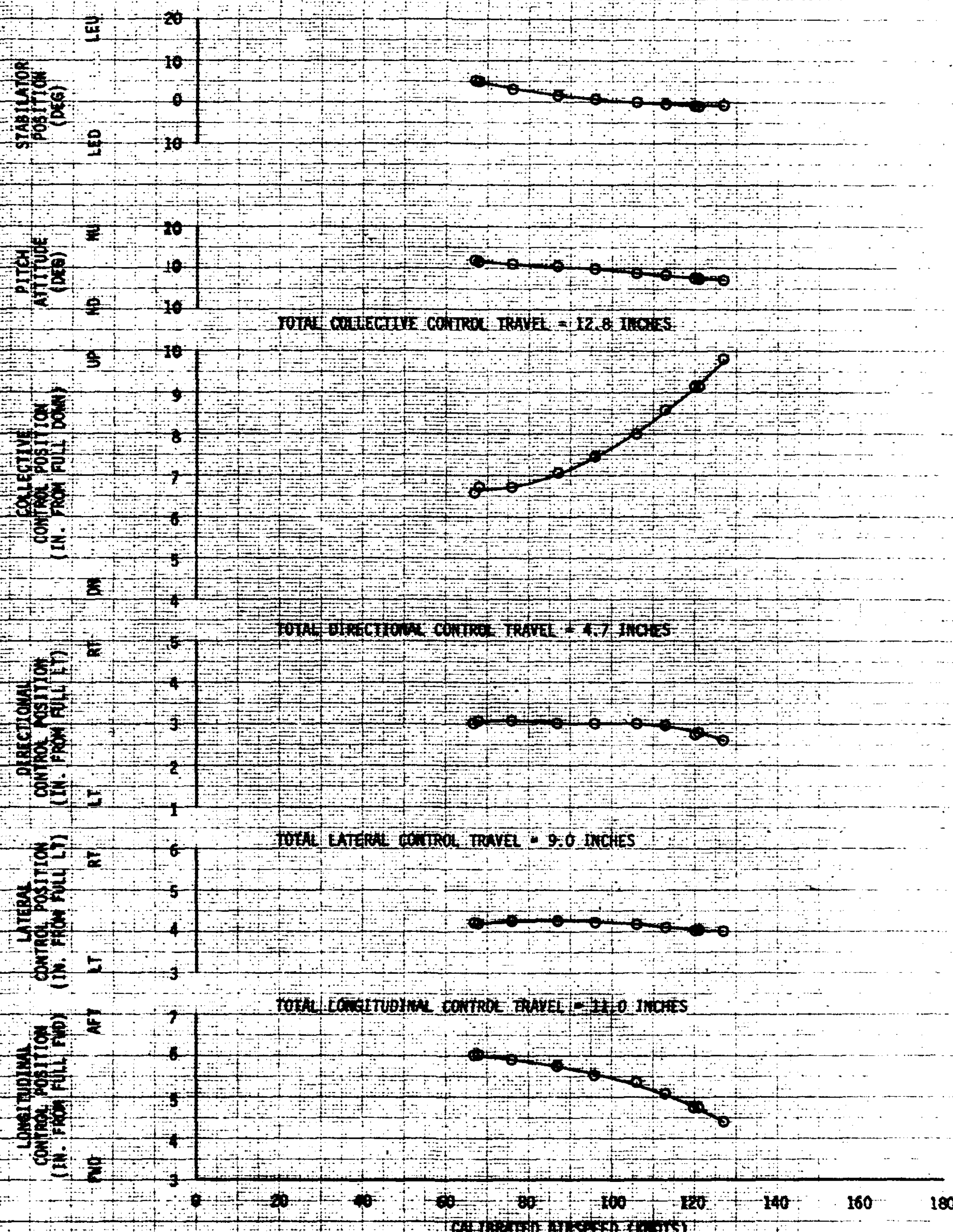


FIGURE 15
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

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	DASE CONDITION
15940	204.3 (AFT)	0.0 (MID)	6600	21	289	LEVEL	ON

NOTES: 1. 16-HELLFIRE CONFIGURATION
2. STABILATOR AUTO MODE
3. TRIM FEEL ON
4. ATTITUDE HOLD OFF
5. SHADED SYMBOLS DENOTE TRIM

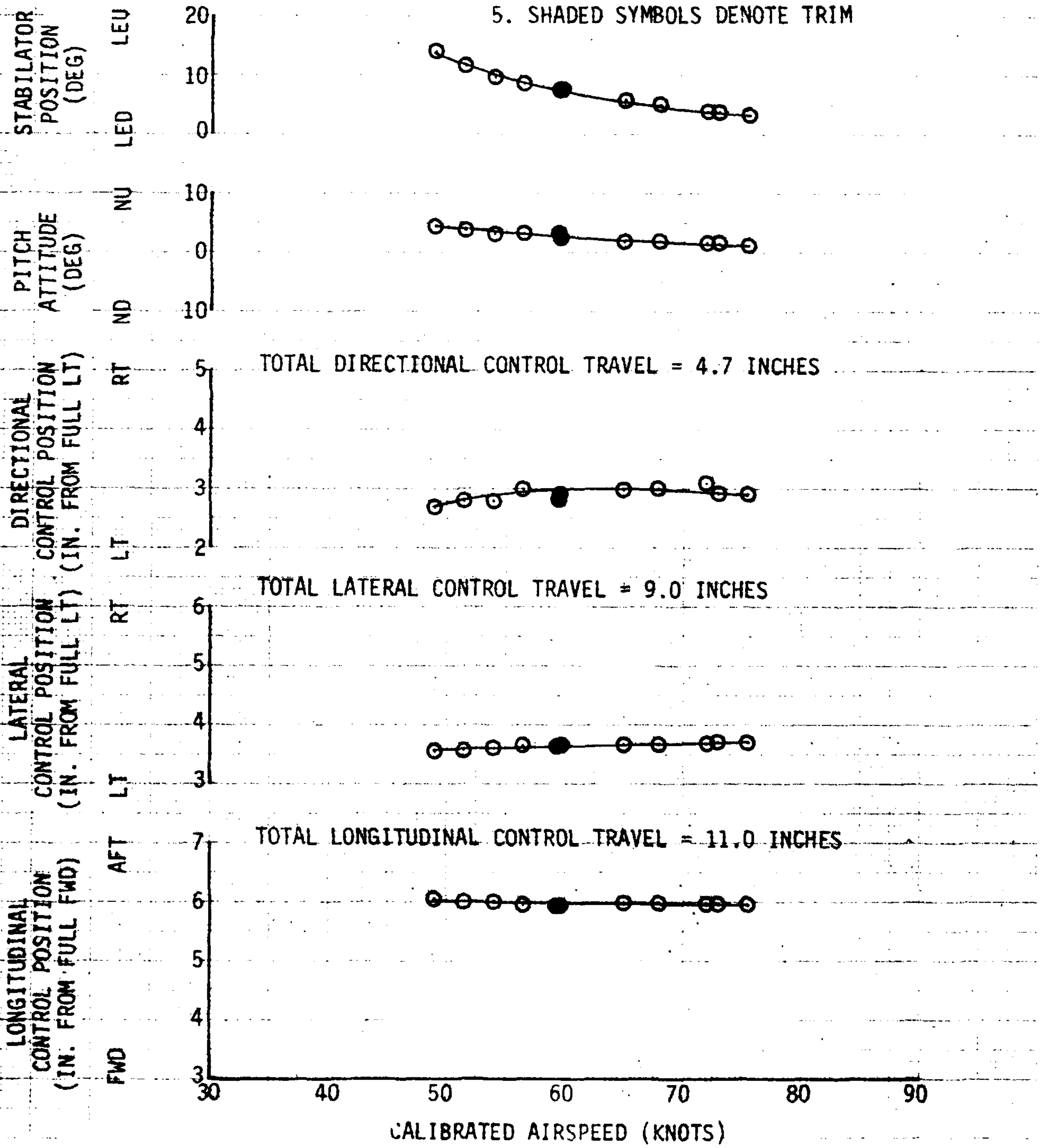


FIGURE 16
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

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	DASE CONDITION
15560	204.4 (AFT)	0.0	6840	20	289	LEVEL	ON

NOTES:
1. 16-HELLFIRE CONFIGURATION
2. STABILATOR AUTO MODE
3. TRIM FEEL ON
4. ATTITUDE HOLD OFF
5. SHADED SYMBOLS DENOTE TRIM

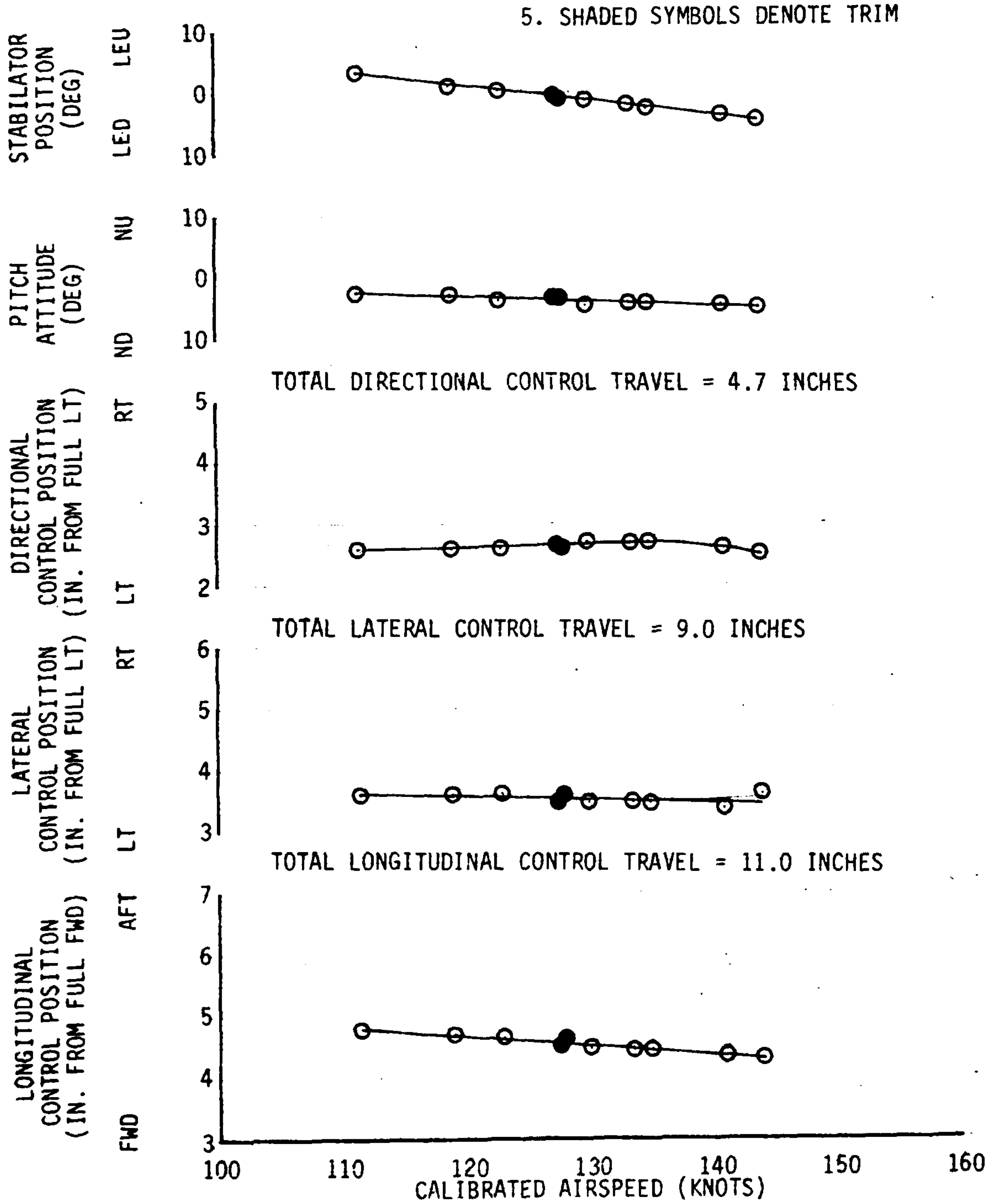


FIGURE 17
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

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)	FLIGHT CONDITION	DASE CONDITION
○	15700	204.4 (AFT)	0.0	7440	18.5	289	RT TURN	ON
□	15260	204.4 (AFT)	0.0	6940	20.0	289	LT TURN	ON

NOTES:
1. 16 HELLFIRE CONFIGURATION
2. 126 KCAS
3. FORCE TRIM ON
4. STABILATOR AUTO MODE
5. ATTITUDE HOLD OFF

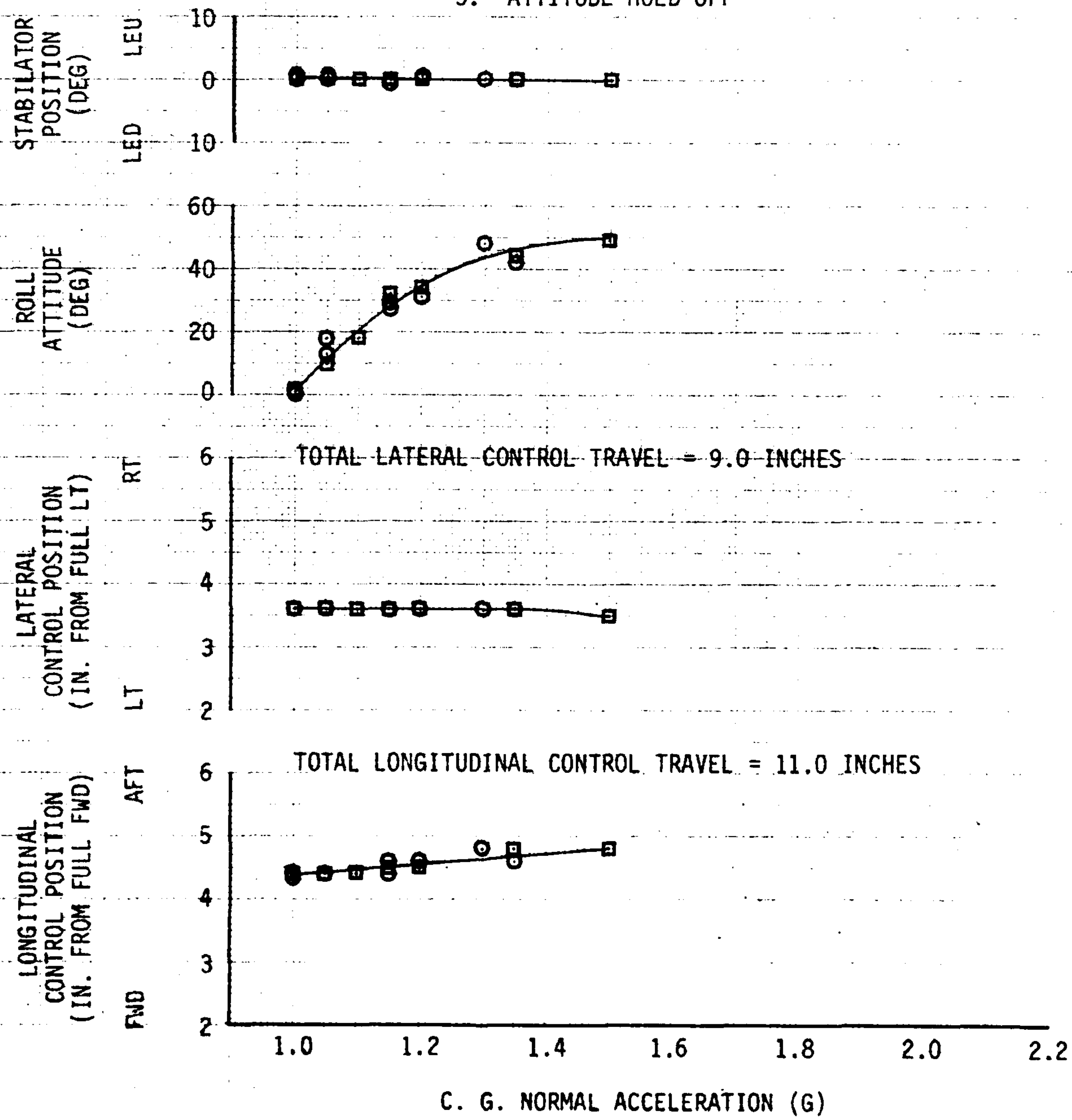


FIGURE 18
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15440	204.4 (FWD)	0.0 (MID)	7460	18.5	289	RIGHT TURN	ON

NOTES:
1. 16-HELLFIRE CONFIGURATION
2. FORCE TRIM ON
3. STABILATOR AUTO MODE
4. ATTITUDE HOLD OFF

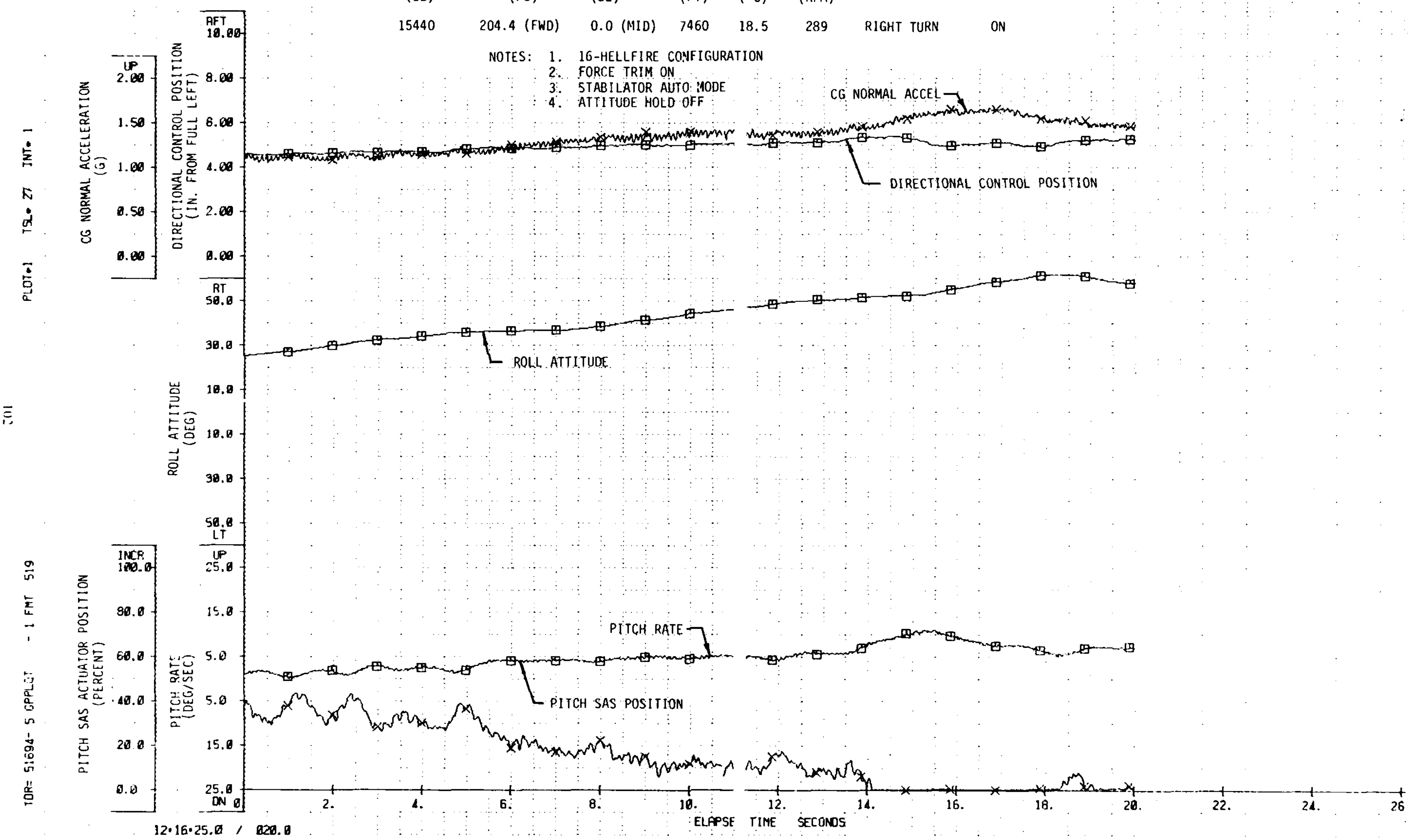


FIGURE 19
DYNAMIC STABILITY
YAW OSCILLATION

YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15300	202.0 (FWD)	0.0 (MID)	4960	28.5	290	LEVEL	ON

NOTE: 8-HELLFIRE CONFIGURATION

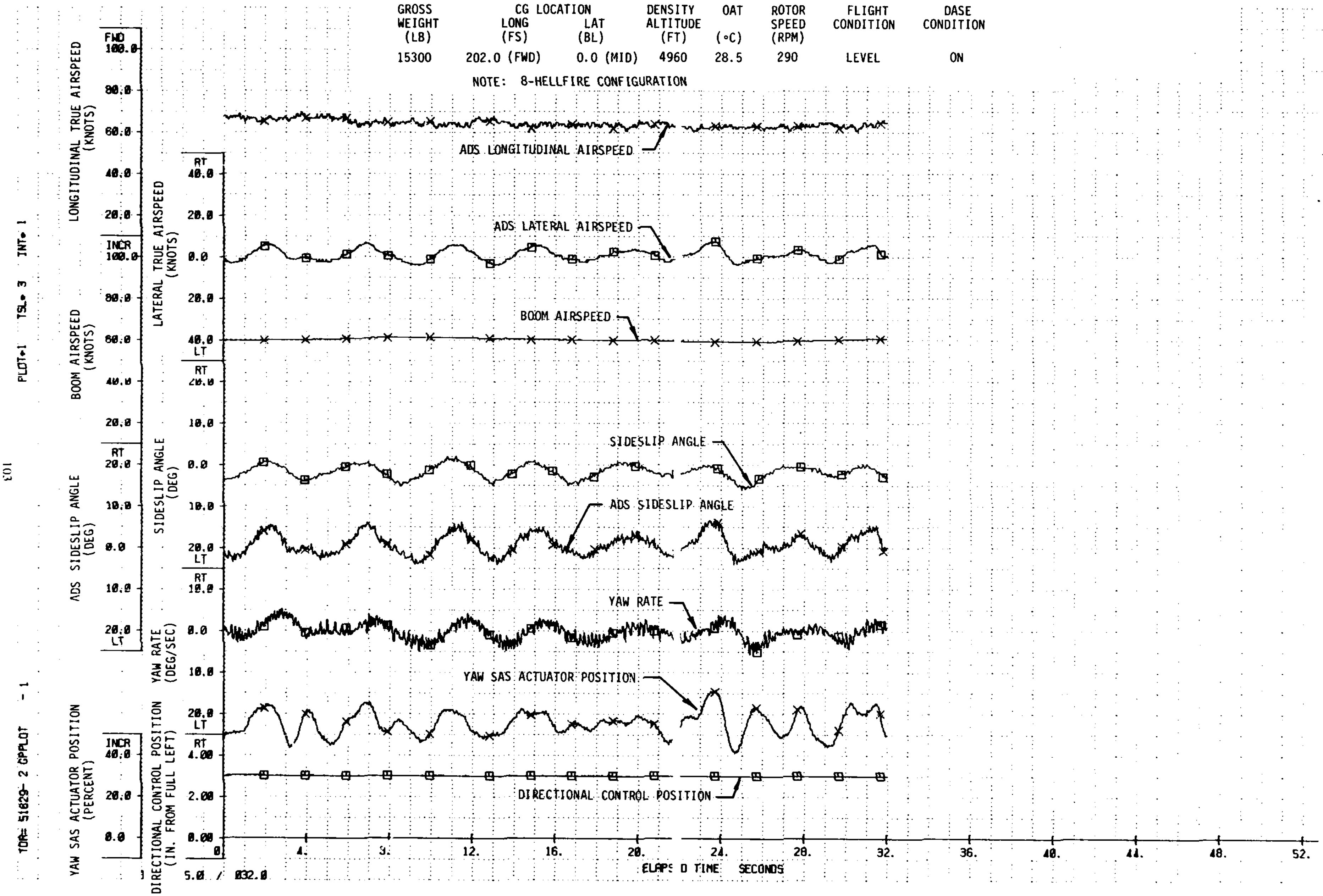


FIGURE 20
DYNAMIC STABILITY
YAW OSCILLATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15400	202.1 (FWD)	0.0 (MID)	3740	22.5	289	LEVEL	YAW SAS OFF

NOTE: 8-HELLFIRE CONFIGURATION

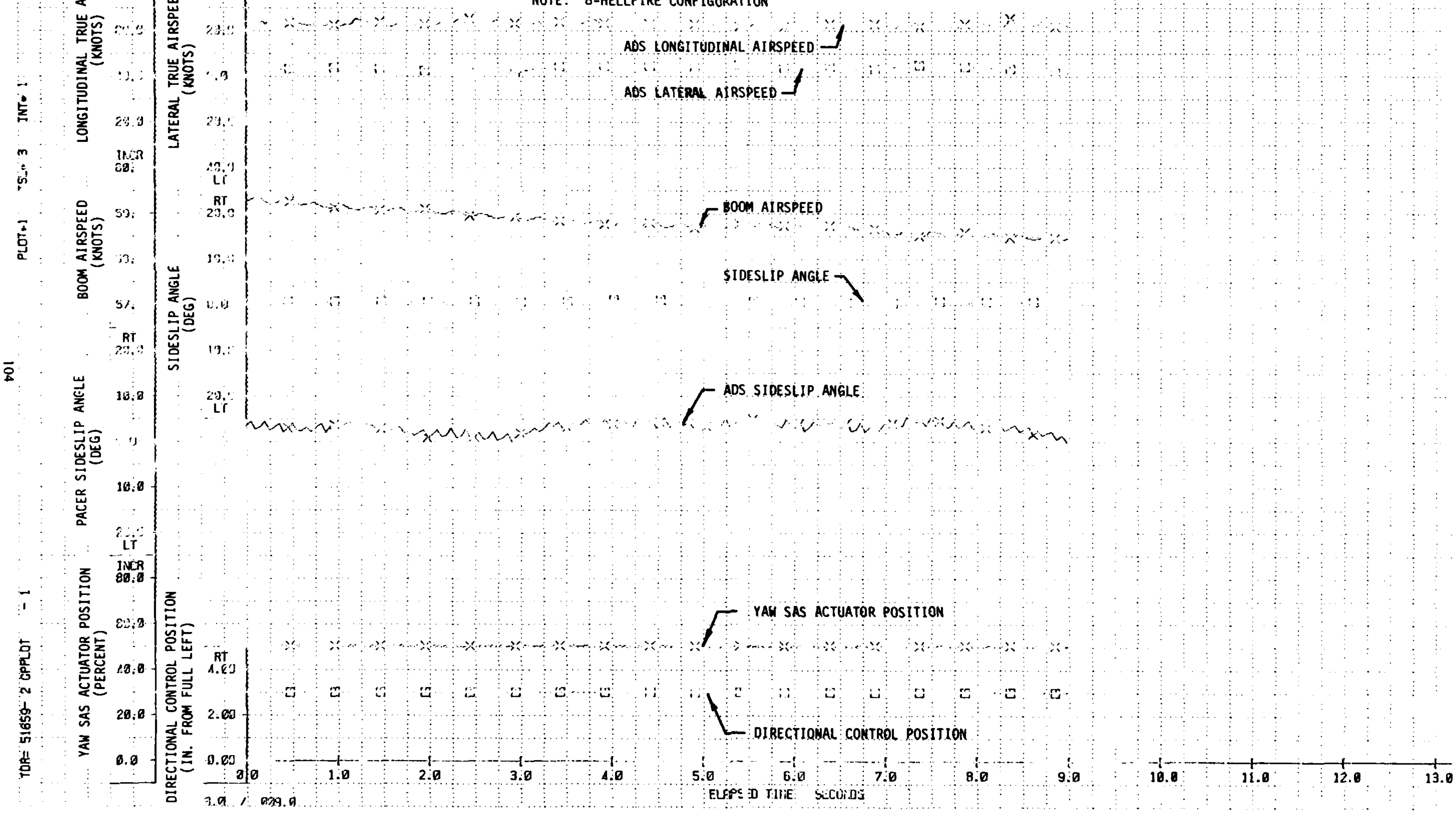


FIGURE 21
DYNAMIC STABILITY
YAW OSCILLATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15220	202.1 (FWD)	0.0 (MID)	4400	25.0	289	LEVEL	ON

NOTE: 1. 8-HELLFIRE CONFIGURATION
2. ADS CIRCUIT BREAKER PULLED

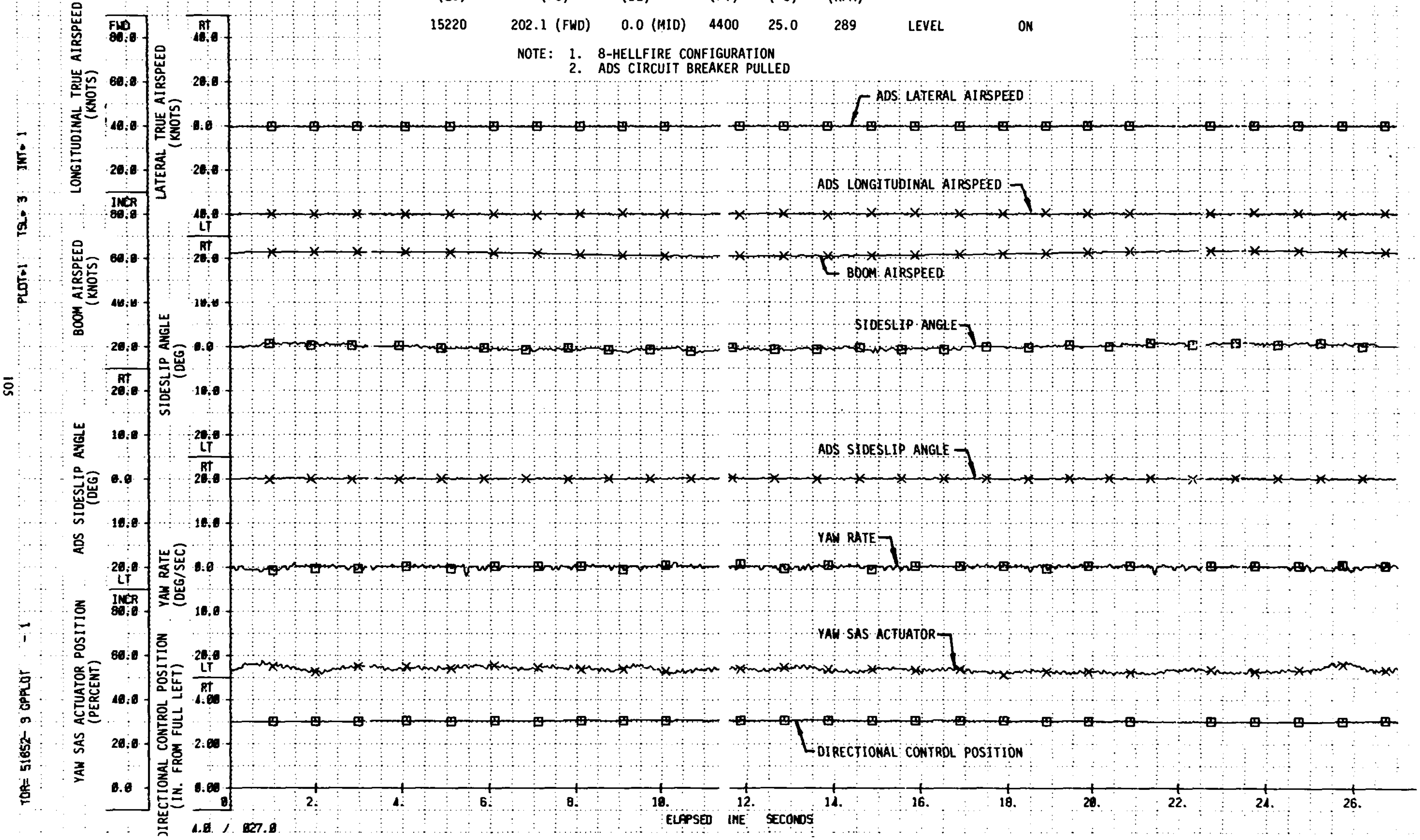


FIGURE 22
LOW SPEED FORWARD AND REARWARD FLIGHT
YAH-64 USA S/N 77-23258

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)	WHEEL HEIGHT (FT)	DASE CONDITION
15240	202.4(Fwd)	0.0 (Mid)	6580	27.0	289	20	ON

NOTES: 1. 8-Hellfire configuration.
2. I denotes extreme travel from trim.
3. Test done in winds of 5-knots or less.
4. Trim feel off

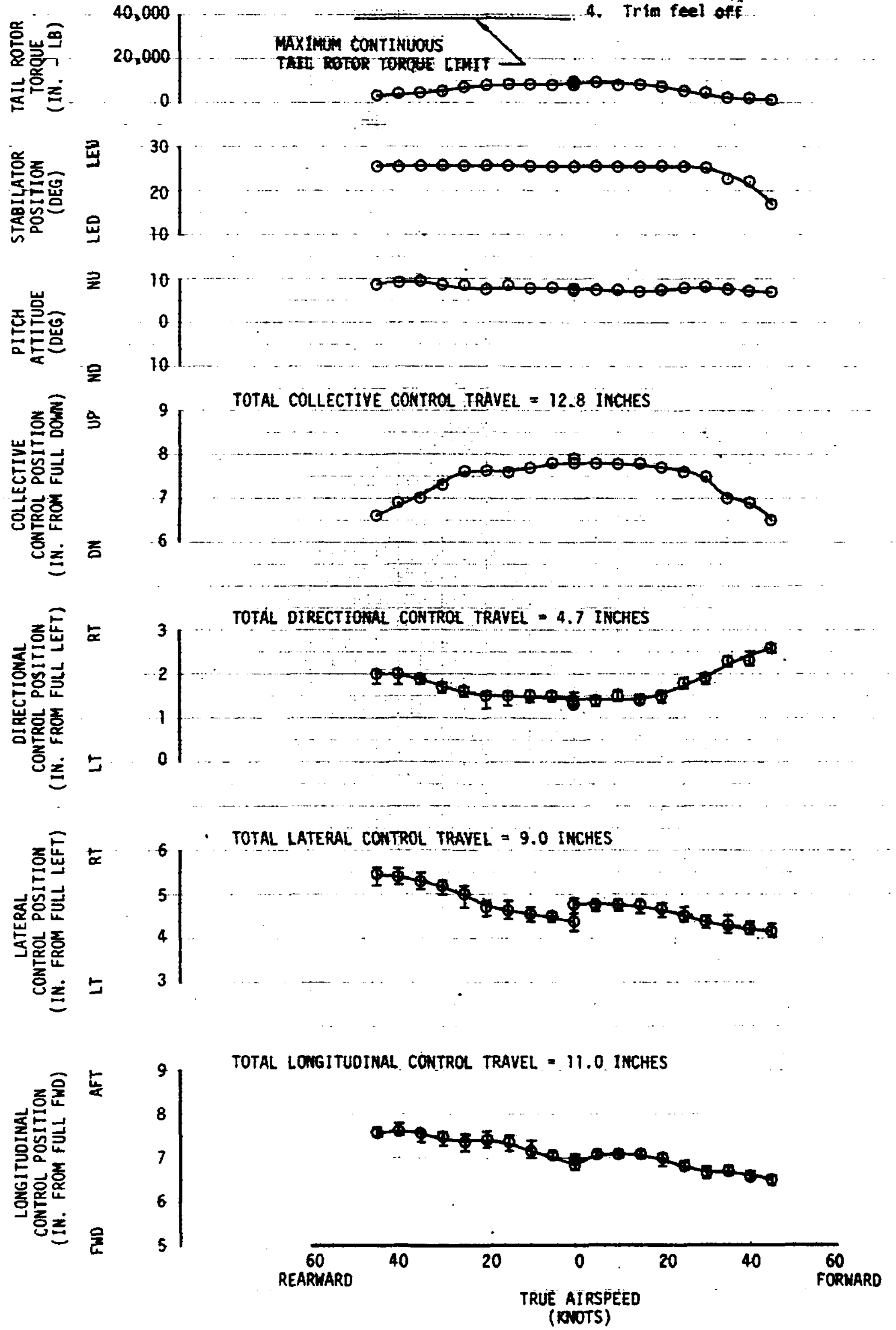


FIGURE 23
LOW-SPEED FORWARD AND REARWARD FLIGHT
YAH-64 USA S/N 77-23258

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)	WHEEL HEIGHT (FT)	DASE CONDITION
15240	202.5 (FWD)	0.0 (MID)	10540	4.5	289	20	ON

NOTES:
1. 8-Hellfire configuration.
2. I denotes extreme travel from trim.
3. Test done in winds of 5-knots or less.
4. Trim feel off

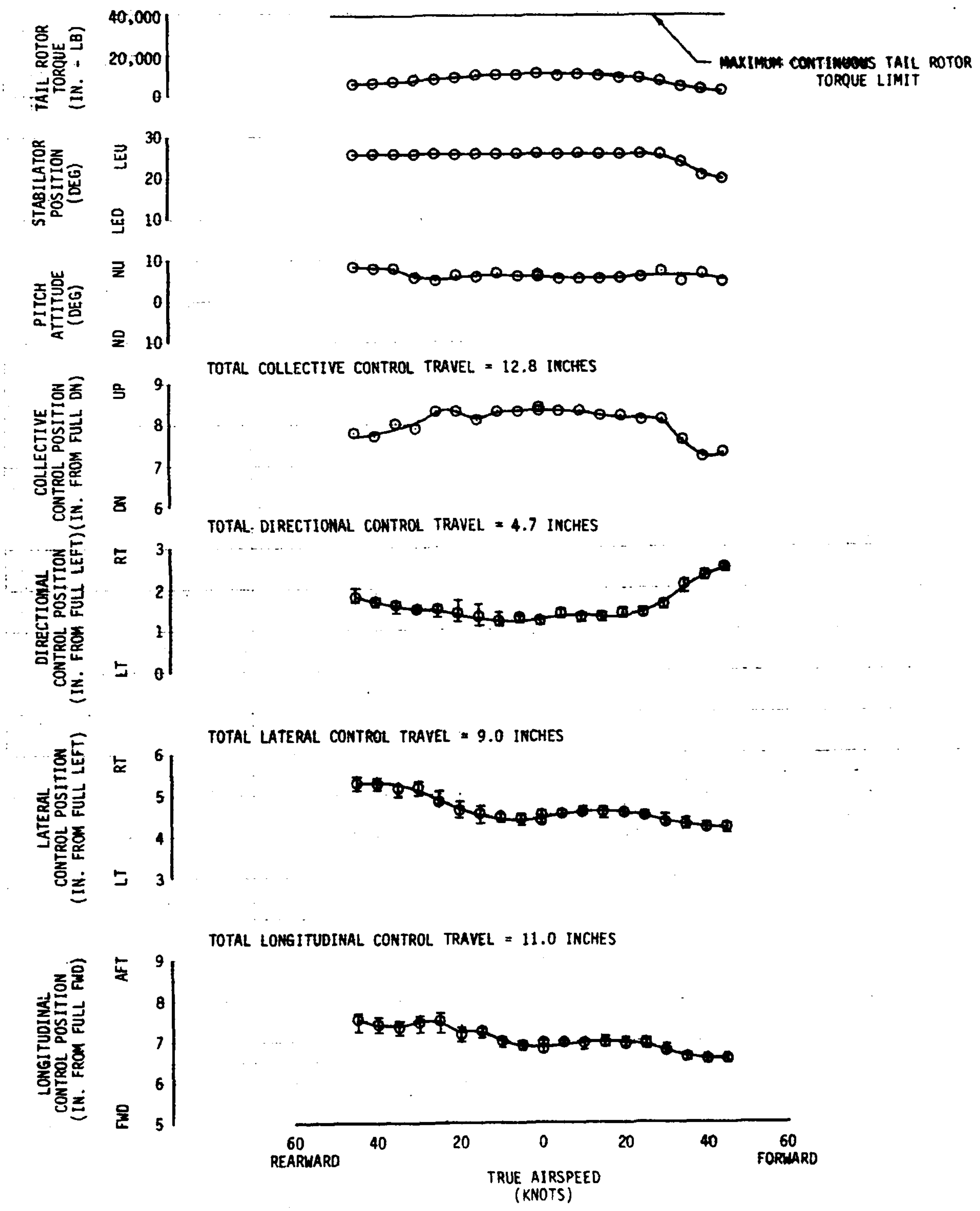


FIGURE 24
LOW SPEED FORWARD AND REARWARD FLIGHT
YAH-64 USA S/N 77-23258

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)	WHEEL HEIGHT (FT)	DASE CONDITION
16560	201.9(Fwd)	0.0(Mid)	5680	21.0	290	20	ON

NOTES:
1. 16-Hellfire configuration.
2. I denotes extreme travel from trim
3. Test done in winds of 5-knots or less
4. Trim feel on.
5. Tail rotor torque not available

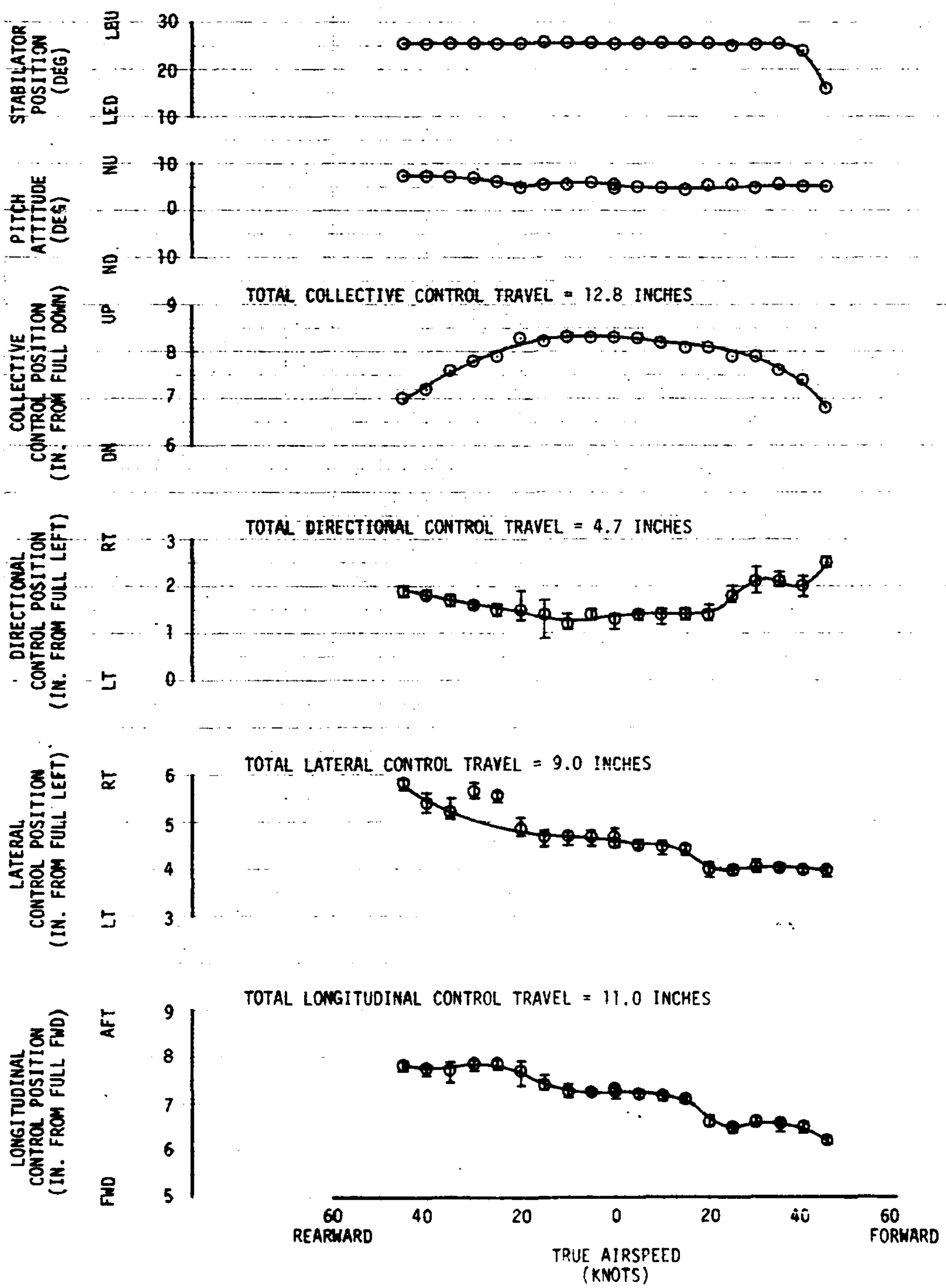


FIGURE 25
SIDEWARD FLIGHT
YAH-64 USA S/N 77-23258

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)	WHEEL HEIGHT (FT)	DASE CONDITION
14960	202.3 (FWD)	0.0 (MID)	6660	27.5	289	20	ON

NOTES: 1. 8-Hellfire configuration.
2. I denotes extreme travel from trim.
3. Test done in winds of 5-knots or less.
4. Trim feel off.

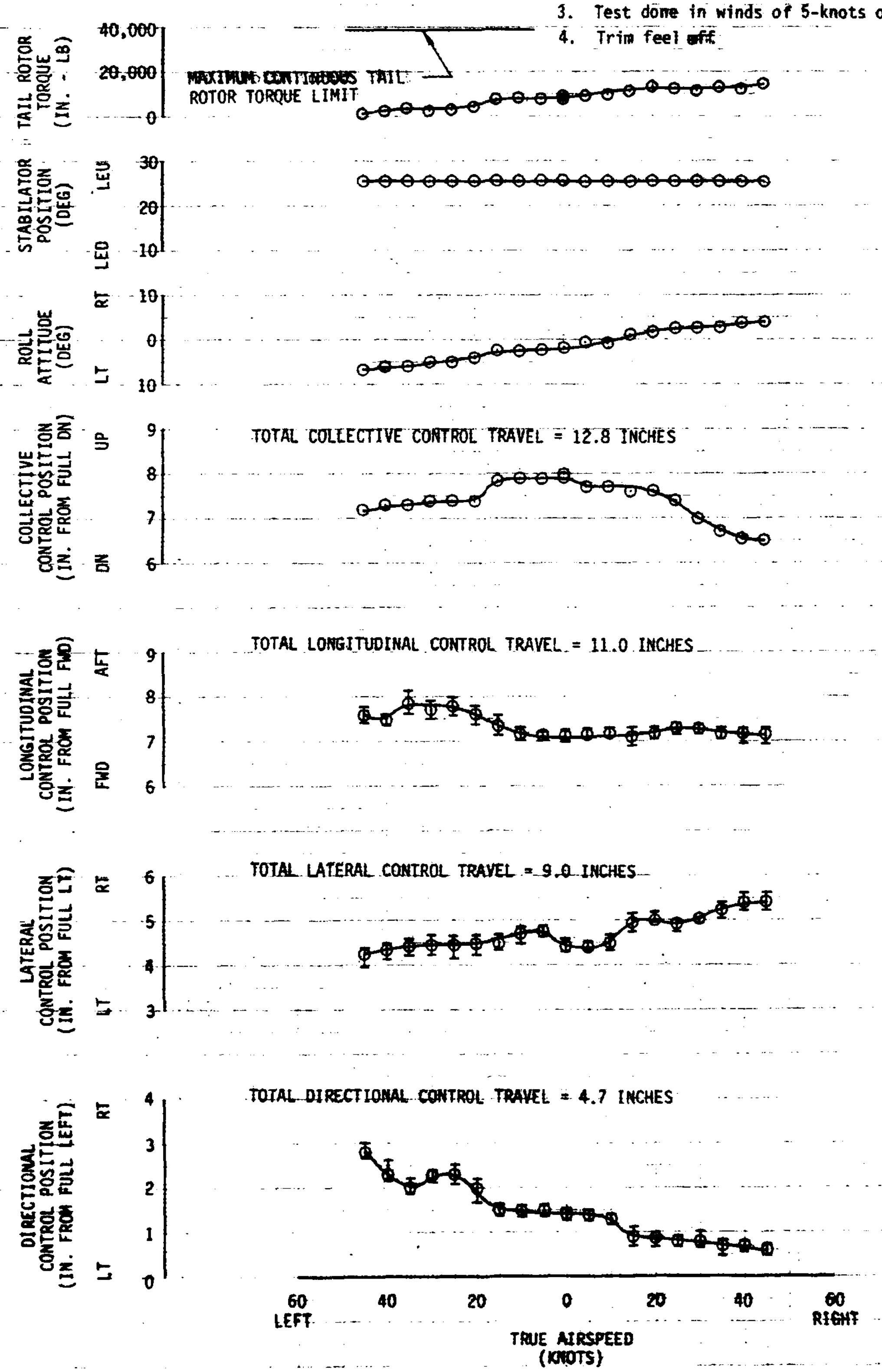


FIGURE 26
SIDEWARD FLIGHT
YAH-64 USA S/N 77-23258

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)	WHEEL HEIGHT (FT)	DASE CONDITION
14440	202.5 (FWD)	0.0 (MID)	10600	4.5	289	20	ON

NOTES:
1. 8-Hellfire configuration.
2. I denotes extreme travel from trim.
3. Test done in winds of 5-knots or less.
4. Trim feel off

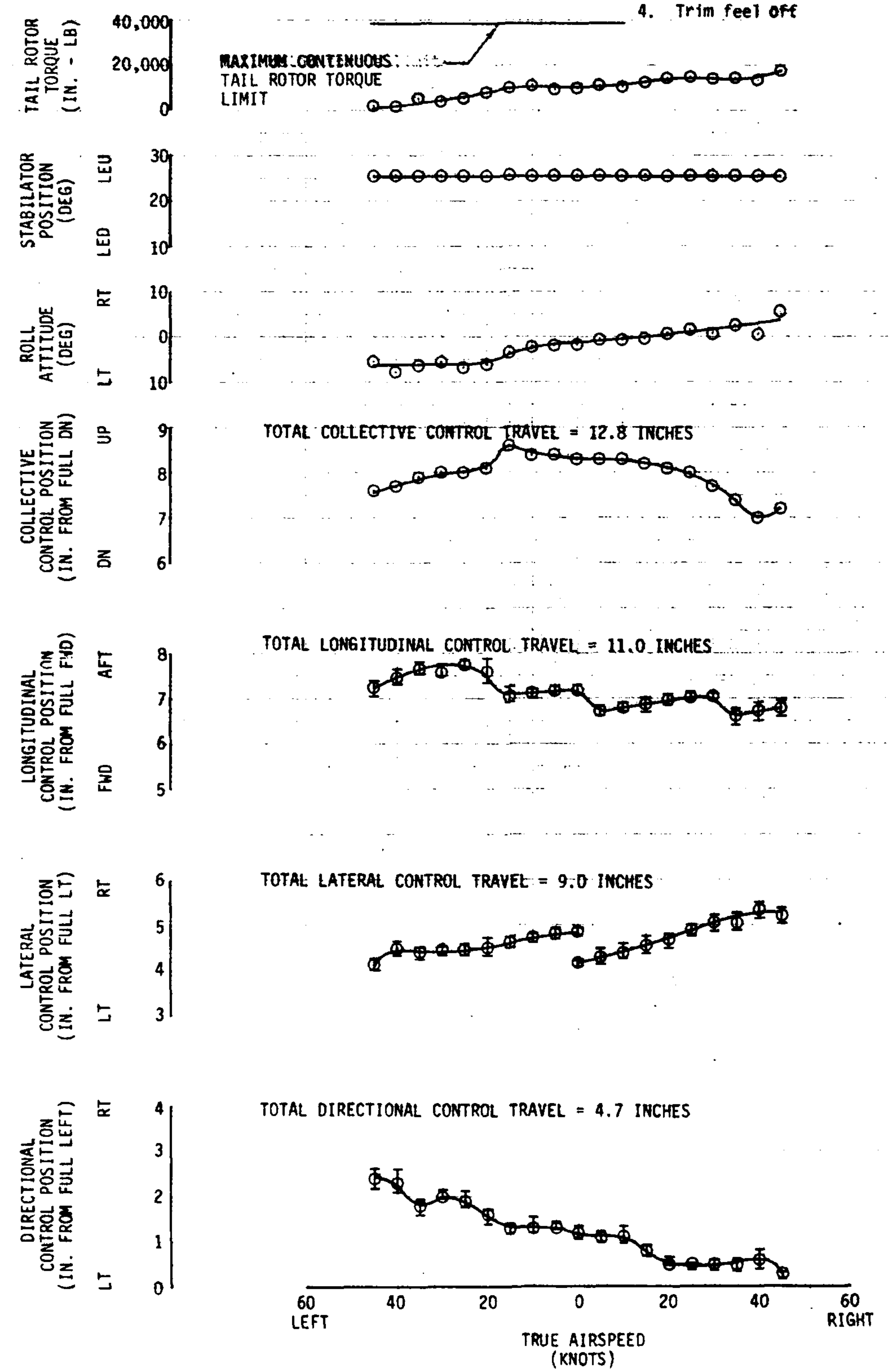
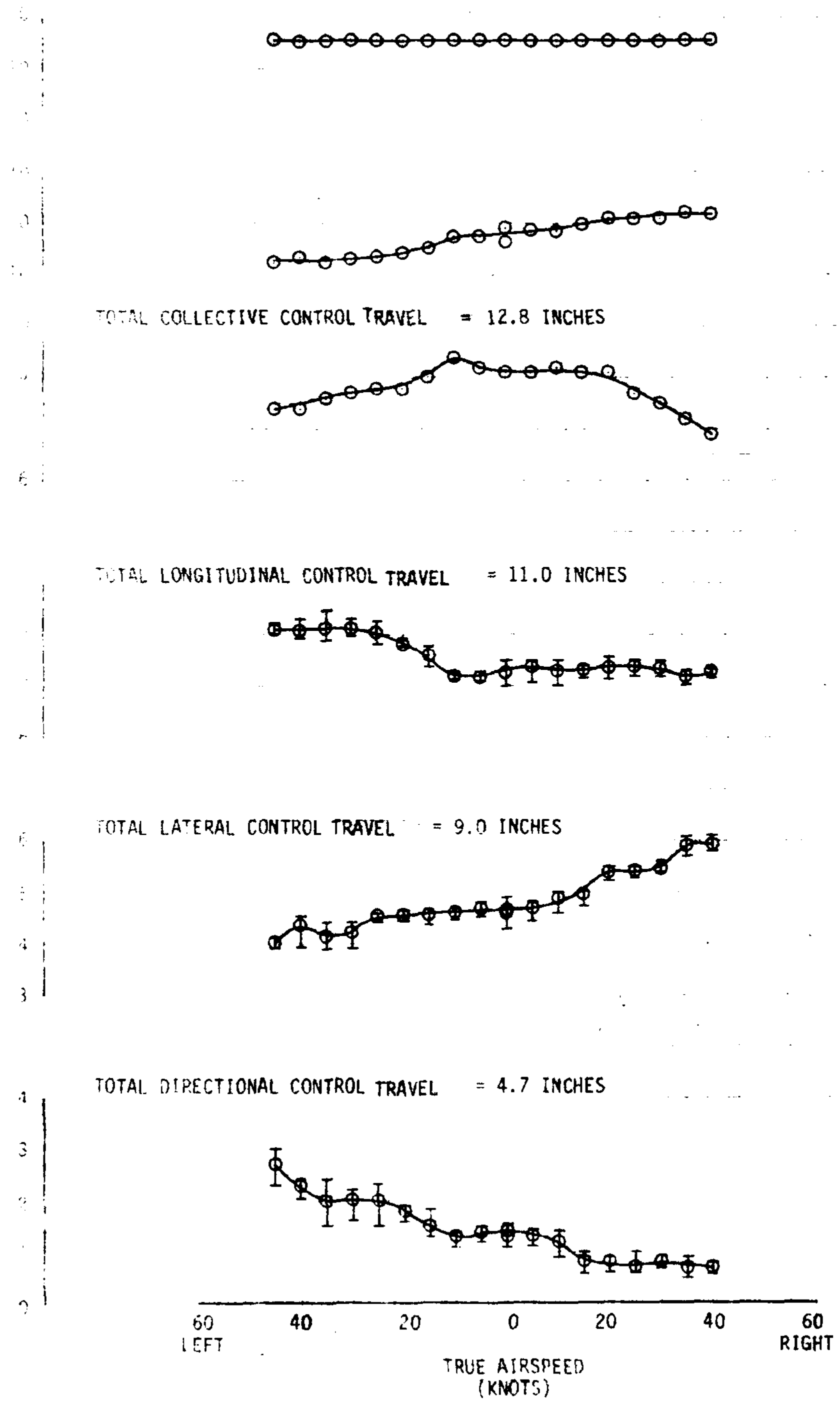
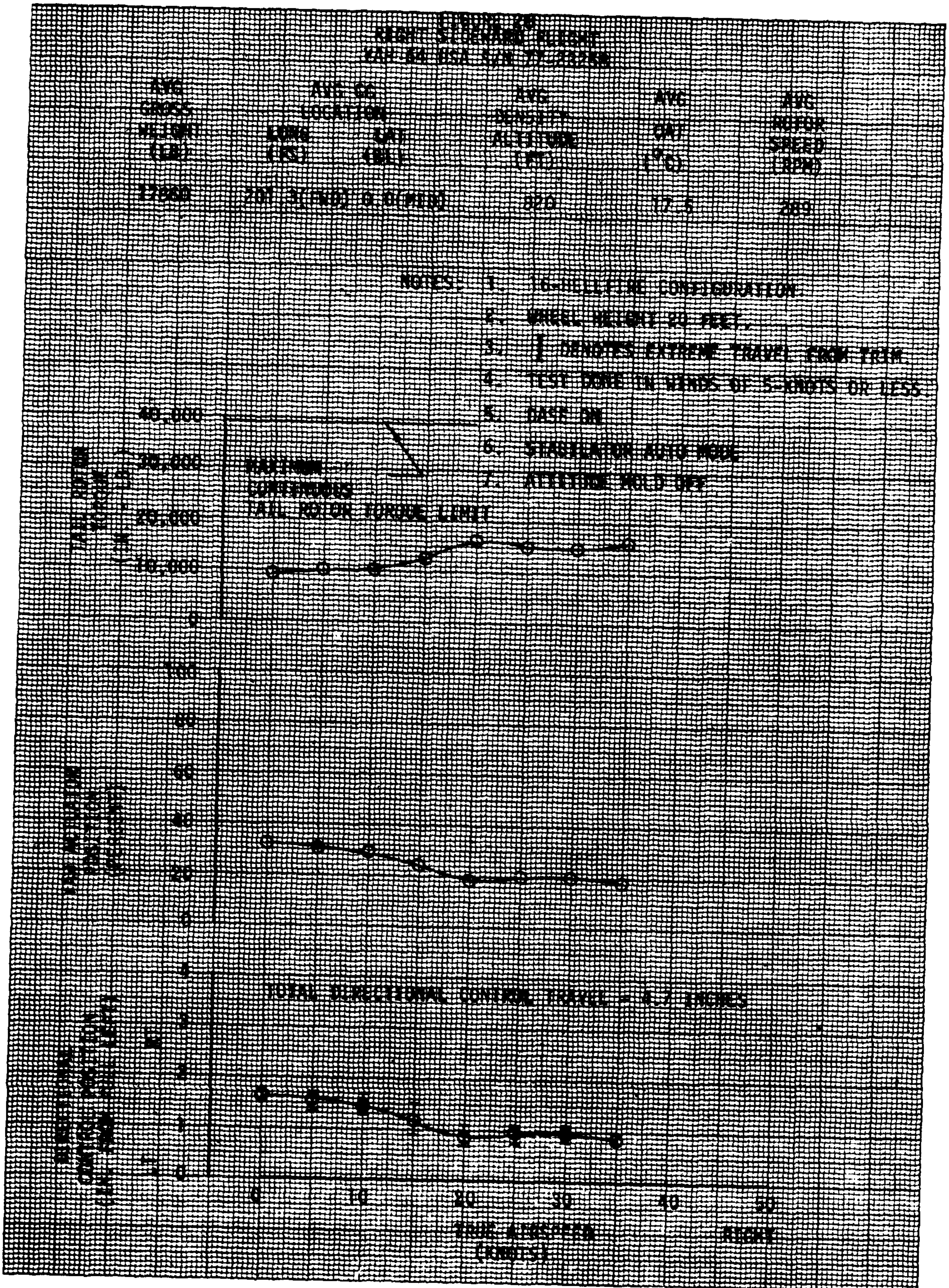


FIGURE 27
SIDEWARD FLIGHT
YAH-64 USA S/N 77-23258

[illegible]	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)	DASE CONDITION
[illegible]	201.8(Fwd)	0.0(Mid)	5800	21.5	290	20	ON

NOTES:
1. 16-Hellfire configuration.
2. I denotes extreme travel from trim.
3. Test done in winds of 5-knots or less.
4. Trim feel on.
5. Tail rotor torque not available



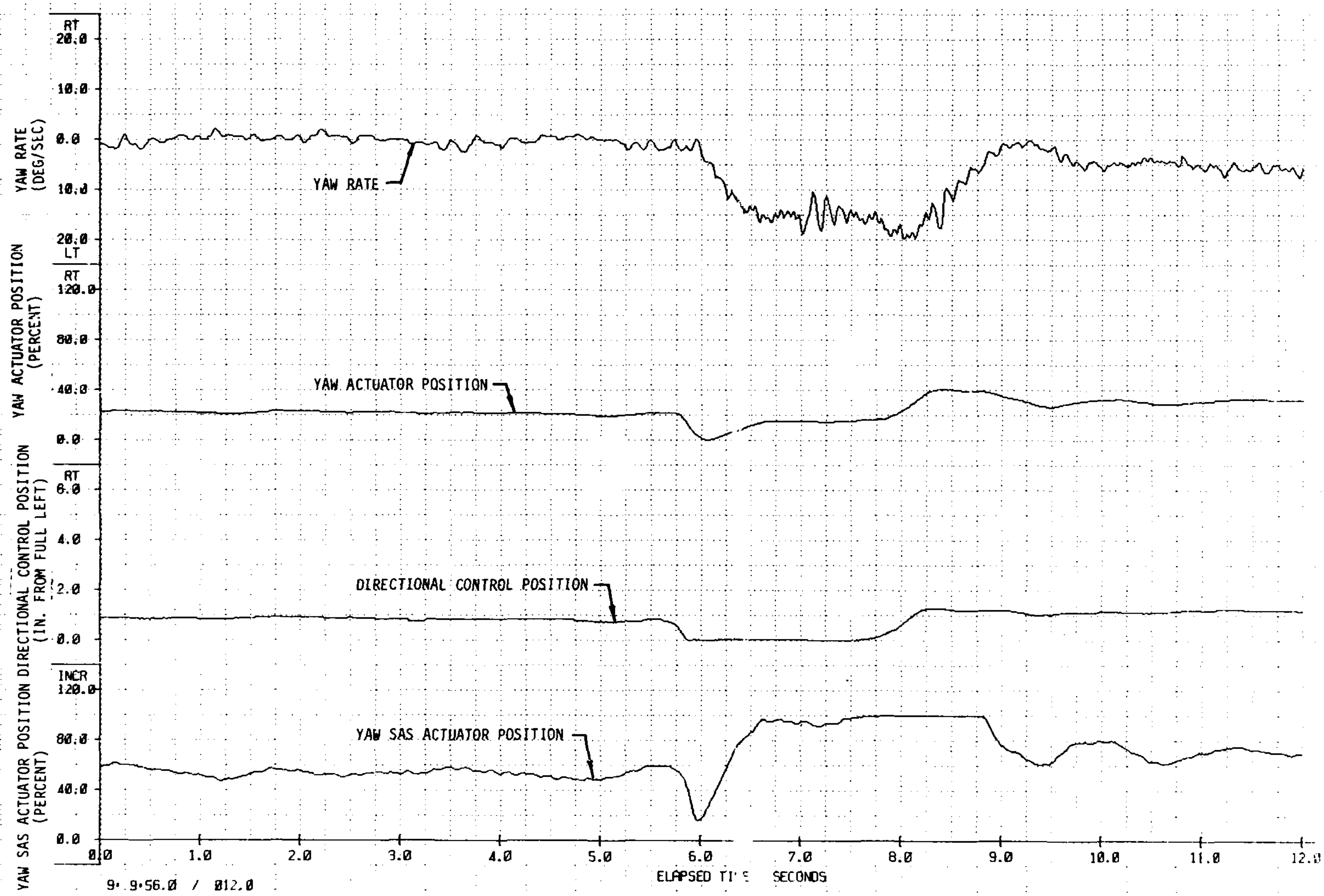


AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
17.5
289
NOTES: 1. 16-HELLFIRE CONFIGURATION.
2. WHEEL HEIGHT 20 FEET.
3. DENOTES EXTREME TRAVEL FROM TRIM.
4. TEST DONE IN WINDS OF 5-KNOTS OR LESS.
5. DASE ON
6. STABILATOR AUTO MODE
7. ATTITUDE HOLD OFF
MAXIMUM CONTINUOUS TAIL ROTOR TORQUE LIMIT
40,000
30,000
20,000
10,000
0
100
80
60
40
20
0
4
3
2
1
0
TOTAL DIRECTIONAL CONTROL TRAVEL = 4.7 INCHES
0
10
20
30
40
50
TRUE AIRSPEED (KNOTS)
RIGHT

FIGURE 29
PEDAL INPUT
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (° C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION	TRUE AIRSPEED (KT)
15,020	202.3 (FWD)	0.0 (MID)	6560	29.0	289	RIGHT SIDEWARD FLIGHT	ON	35

NOTE: 8-HELLFIRE CONFIGURATION



TDR= 51598- 1 CPPLOT - 1 PLOT= 1 TSL= 7 INT= 1

FIGURE 30
LOW-SPEED FORWARD AND REARWARD FLIGHT
YAH-64 USA S/N 77-23258

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)	WHEEL HEIGHT (FT)	DASE CONDITION
15180	202.5 (FWD)	0.0(MID)	720	16.5	289	20	ON

NOTES:
1. Stabilator fixed 25° leading edge up.
2. 8-Hellfire configuration.
3. I denotes extreme travel from trim.
4. Test done in winds of 5-knots or less.
5. Trim feel off.

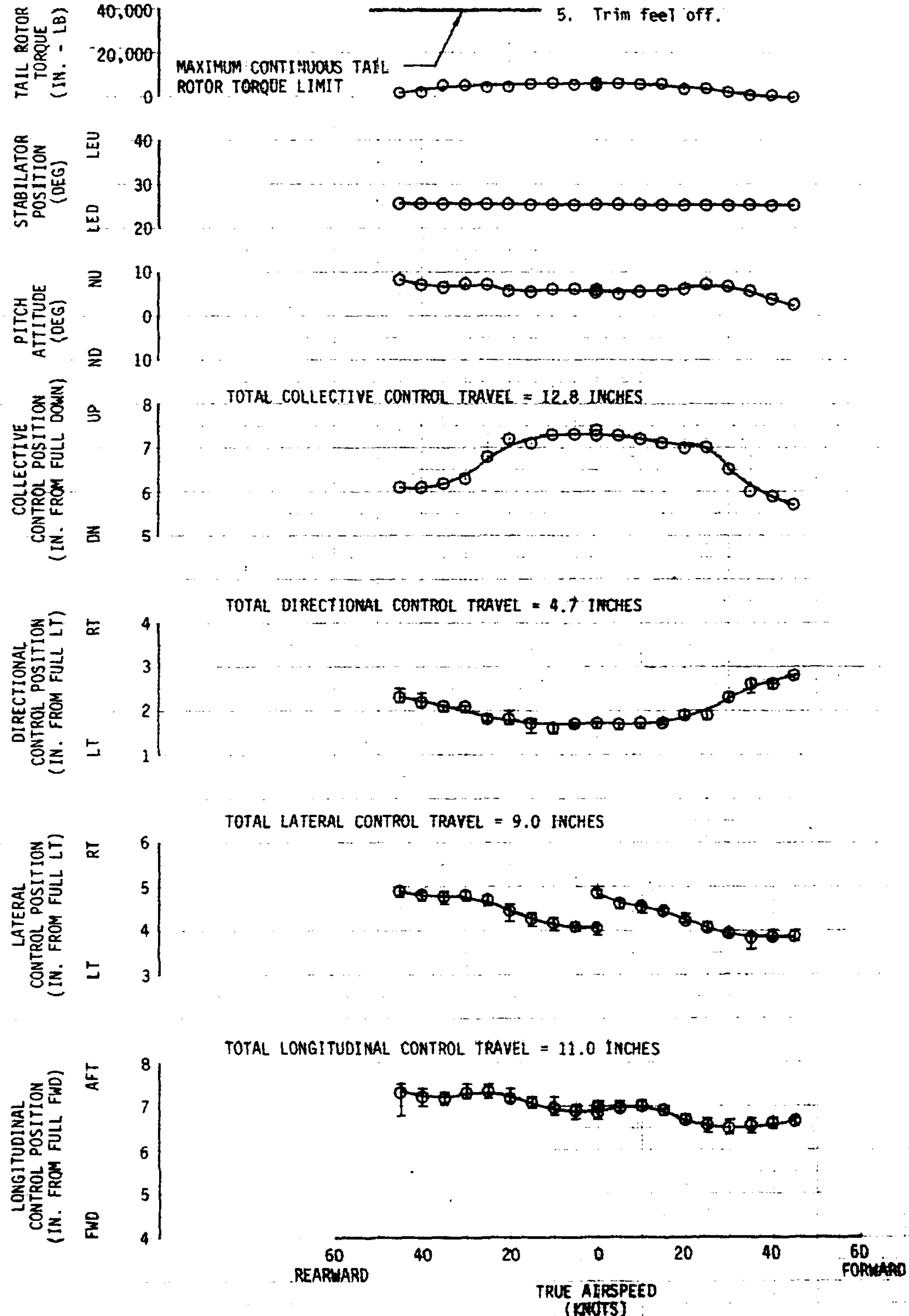


FIGURE 31
LOW-SPEED FORWARD AND REARWARD FLIGHT
YAH-64 USA S/N 77-23258

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)	WHEEL HEIGHT (FT)	BASE CONDITION
14720	202.5 (FWD)	0.0(MID)	700	16.5	289	20	ON

NOTES:
1. Stabilator fixed 35° leading edge up.
2. 8-Hellfire configuration.
3. I denotes extreme travel from trim.
4. Test done in winds of 5-knots or less.
5. Trim feel off.

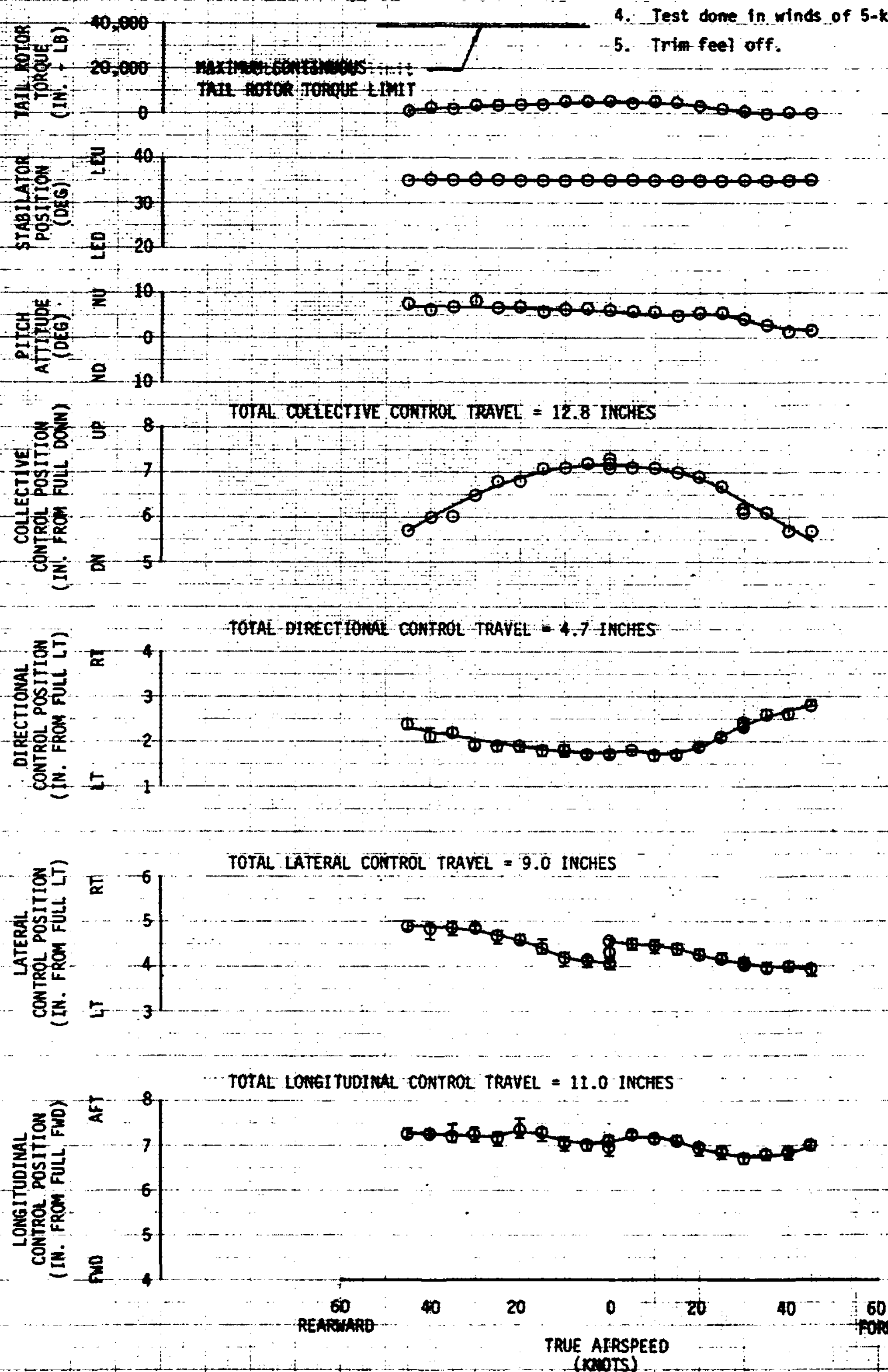


FIGURE 32
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

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
15180	202.1(FWD)	0.0 (MID)	8000	16.0	290	LEVEL

NOTE: 8 HELLFIRE CONFIGURATION
□ STABILATOR FIXED 35 DEGREES LEADING EDGE UP
○ STABILATOR FIXED 25 DEGREES LEADING EDGE UP

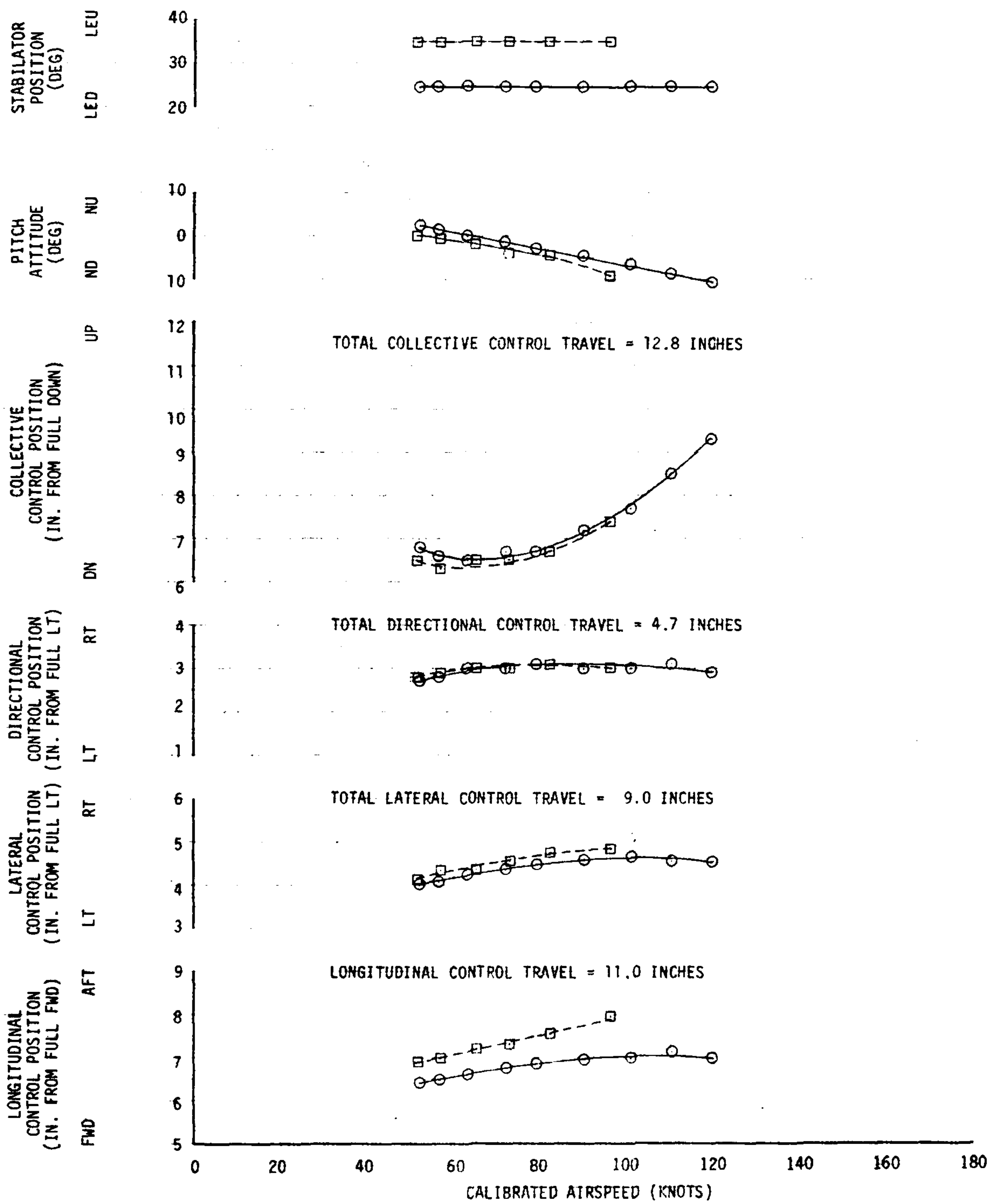


FIGURE 33
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

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	DASE CONDITION
14800	202.1 (FWD)	0.0 (MID)	7940	17.0	289	LEVEL	ON

NOTES: 1. STABILATOR IN NOE/APPROACH MODE
2. 8-HELLFIRE CONFIGURATION
3. SHADED SYMBOLS DENOTE TRIM

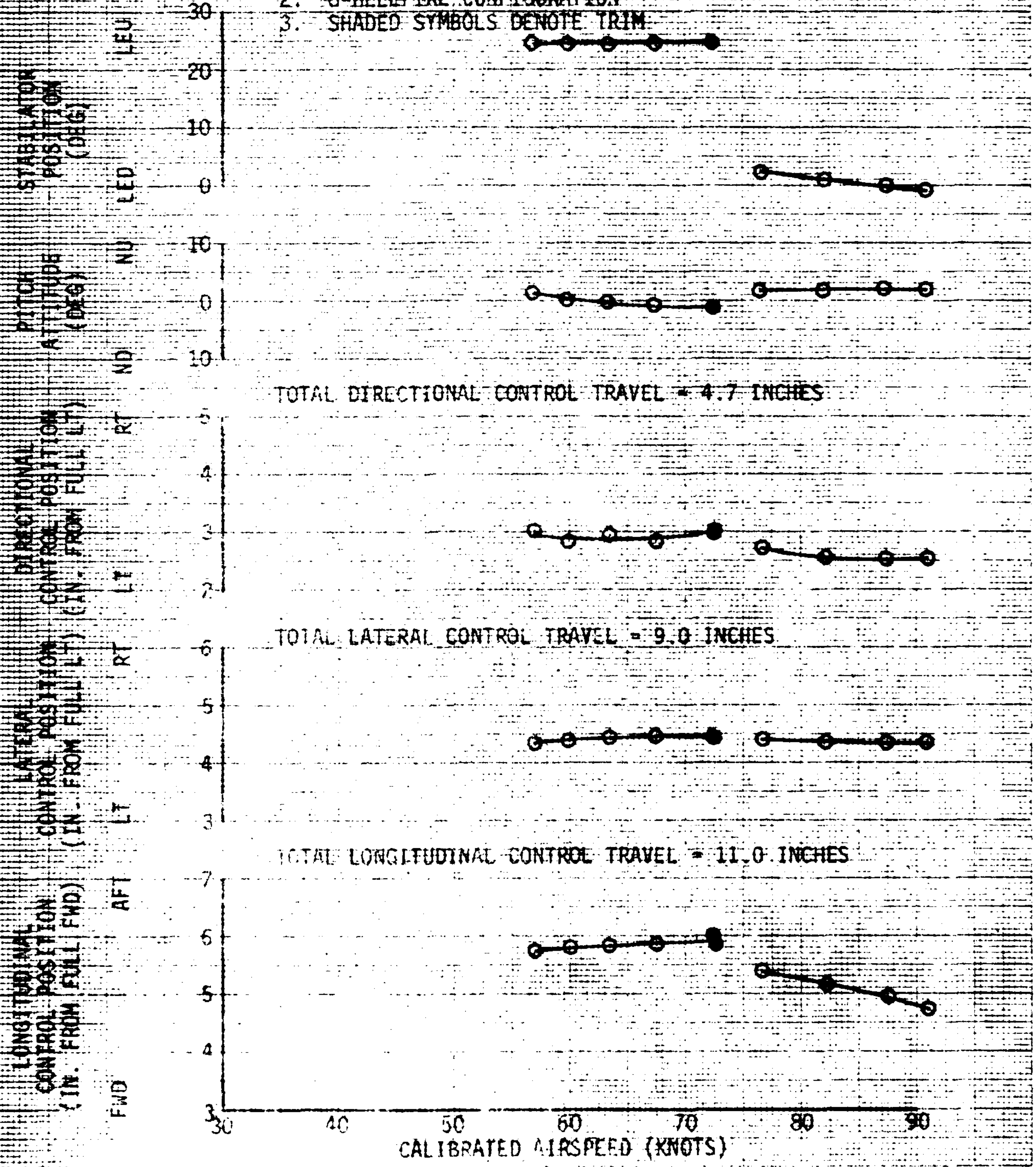
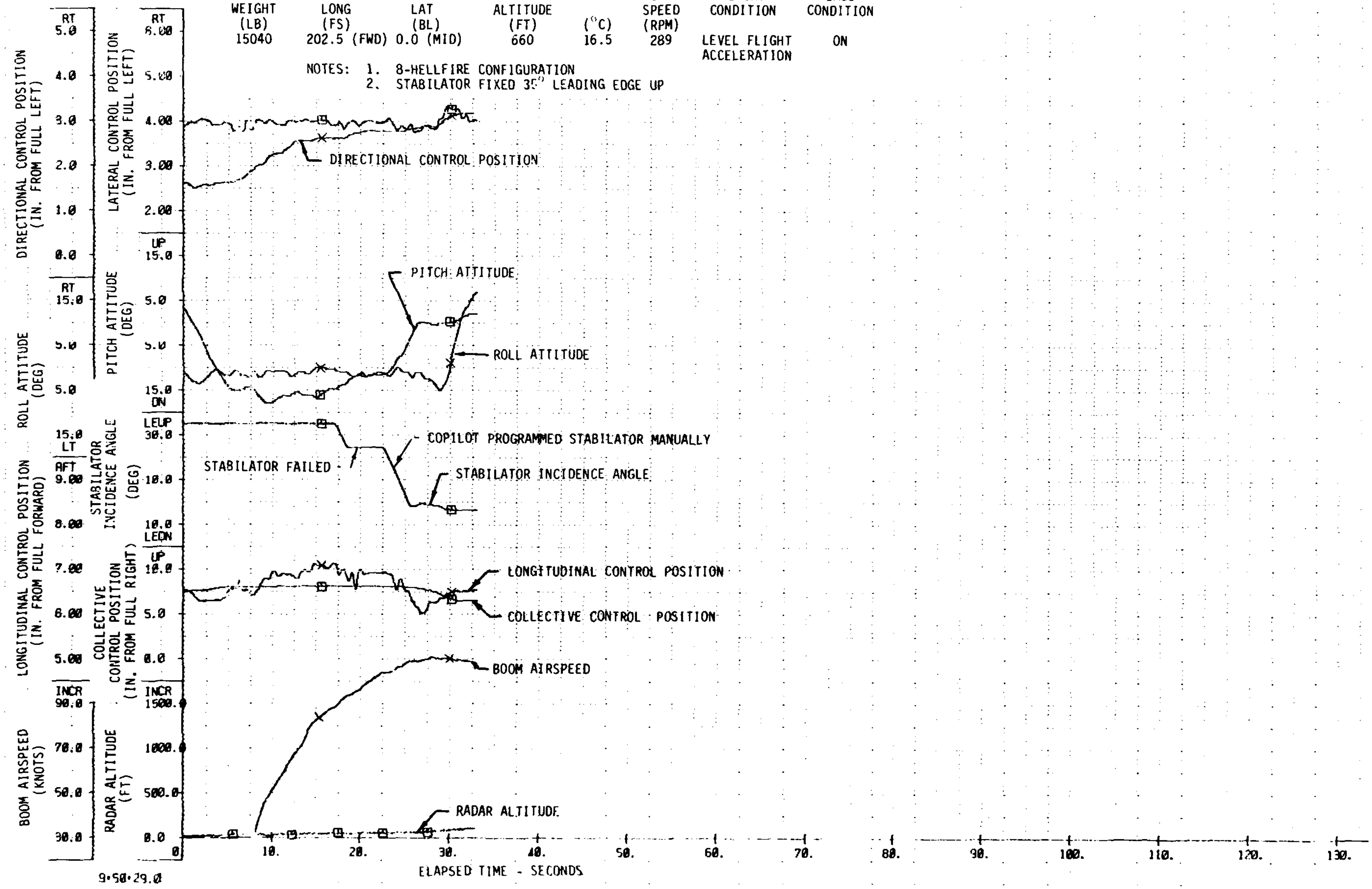


FIGURE 34
STABILATOR EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15040	202.5 (FWD)	0.0 (MID)	660	16.5	289	LEVEL FLIGHT ACCELERATION	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. STABILATOR FIXED 35° LEADING EDGE UP



TDR= 51389- 7 GPPLOT - 1 FMT 514 PLOT•1 TSL• 7 INT• 1

FIGURE 35
STABILATOR EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15220	202.5 (FWD)	0.0 (MID)	620	16.0	291	LEVEL FLIGHT ACCELERATION/ DECELERATION	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. NOE/APP MODE

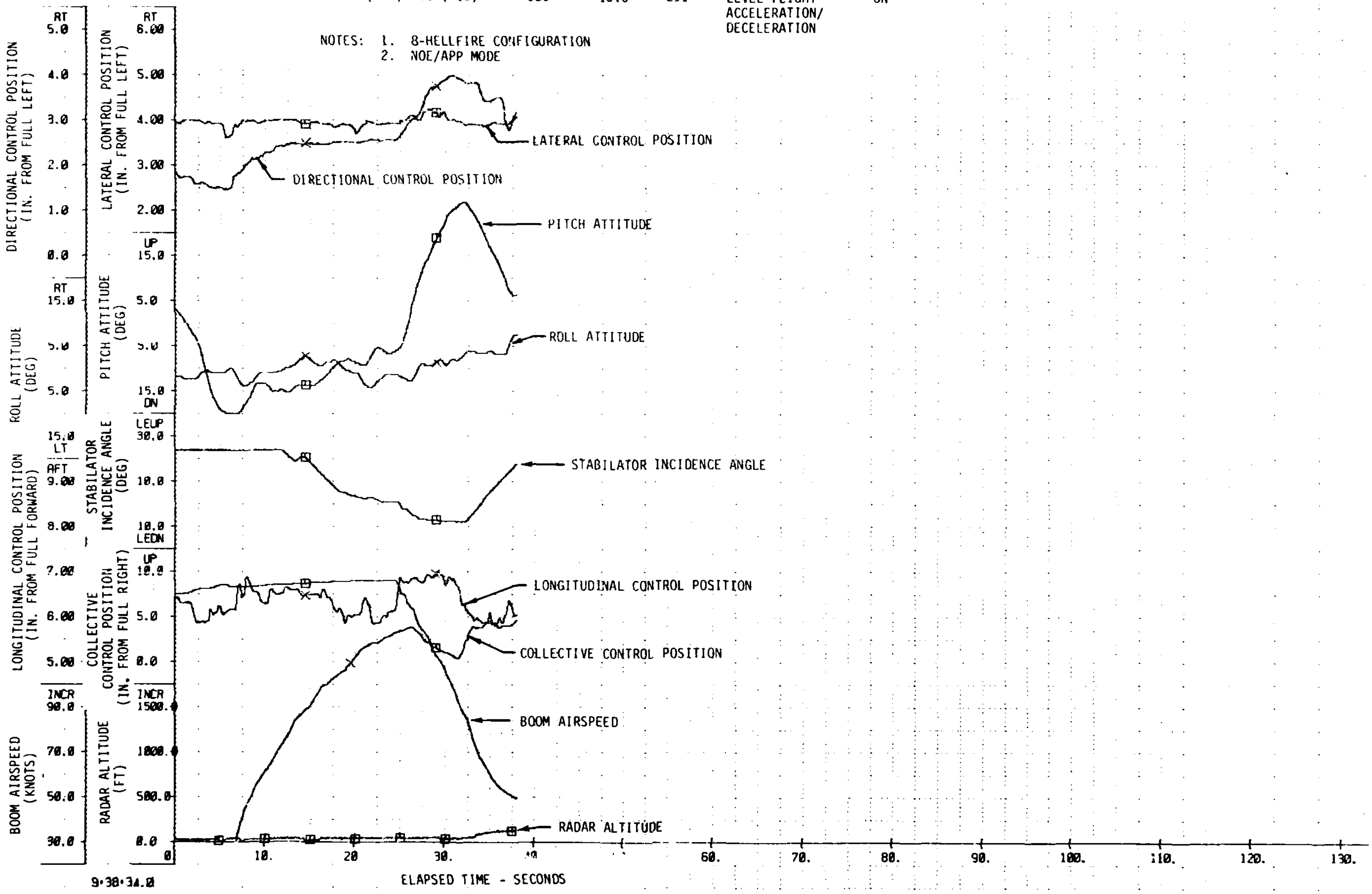
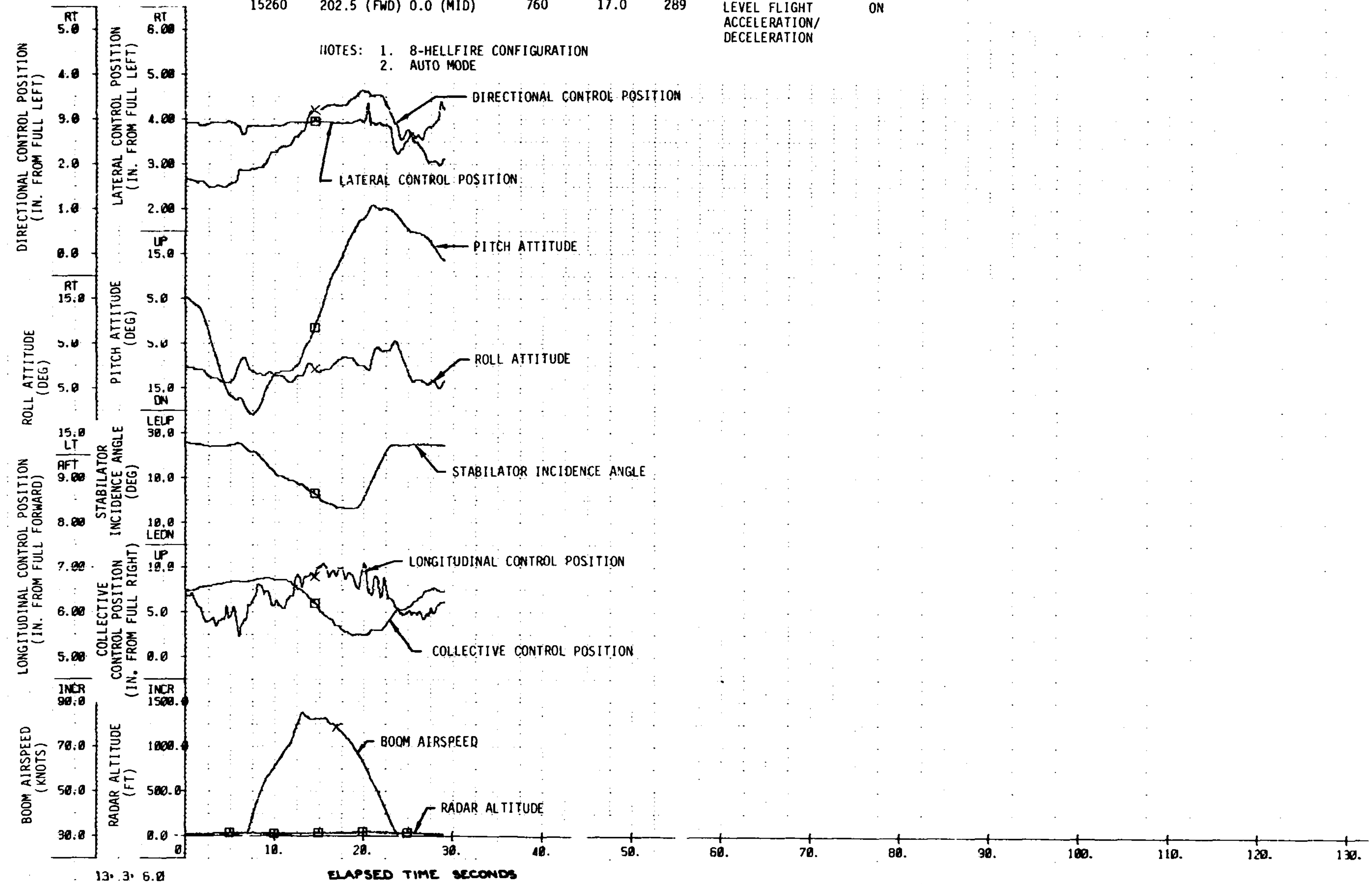


FIGURE 36
STABILATOR EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15260	202.5 (FWD)	0.0 (MID)	760	17.0	289	LEVEL FLIGHT ACCELERATION/ DECELERATION	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. AUTO MODE

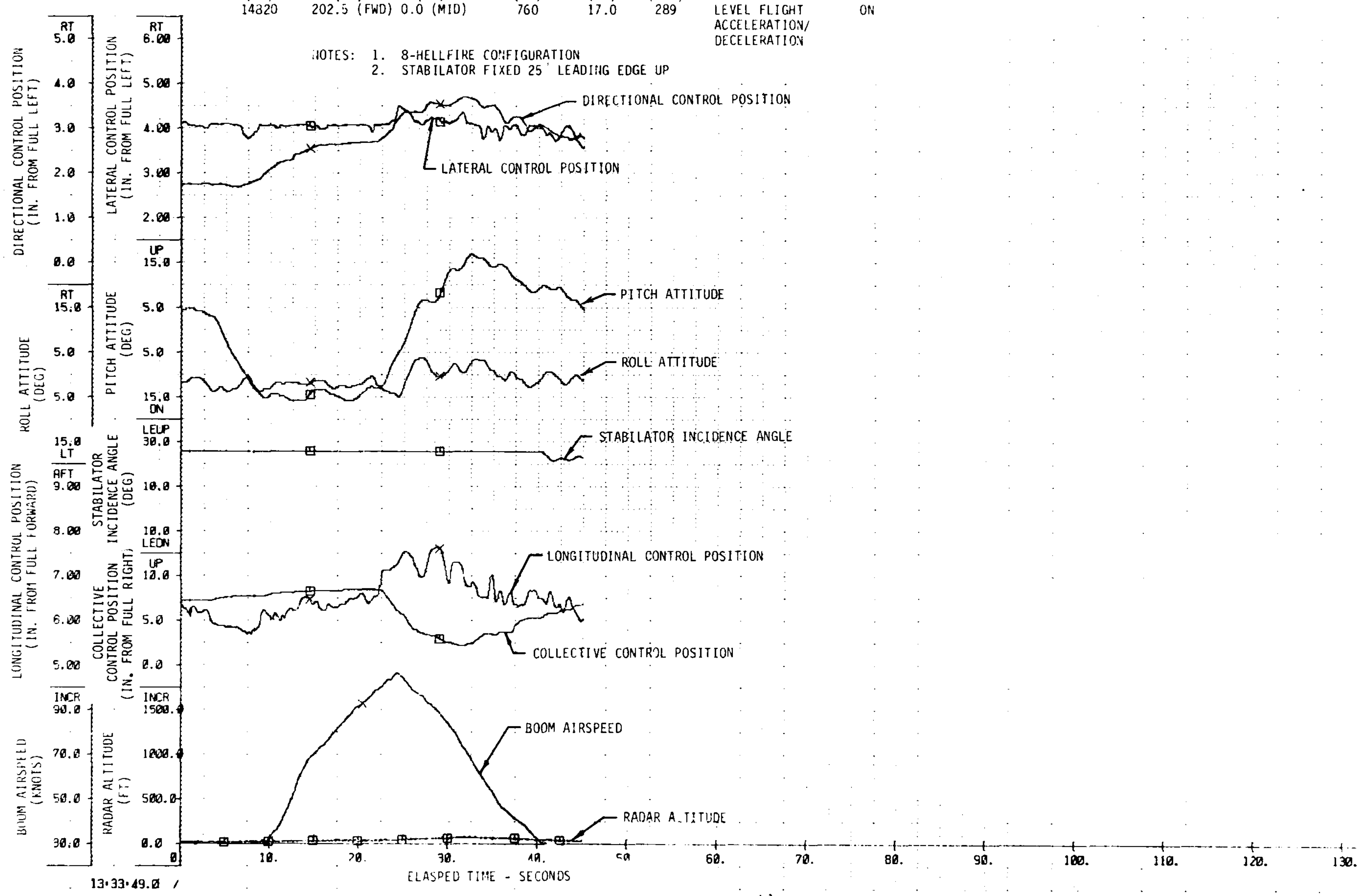


TOR= 51398- 1 GPPLOT - 1 FMT 514 PLOT• 1 TSL• 1 INT• 1

FIGURE 37
STABILATOR EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
14820	202.5 (FWD)	0.0 (MID)	760	17.0	289	LEVEL FLIGHT ACCELERATION/ DECELERATION	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. STABILATOR FIXED 25° LEADING EDGE UP



TOR= 51366- 8 CPPLOT - 1 FMT 514 PLOT• 1 TSL• 8 INT• 1

FIGURE 38
STABILATOR EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
14380	202.5 (FWD)	0.0 (MID)	800	17.5	289	LEVEL FLIGHT ACCELERATION/ DECELERATION	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. STABILATOR FIXED 35° LEADING EDGE UP

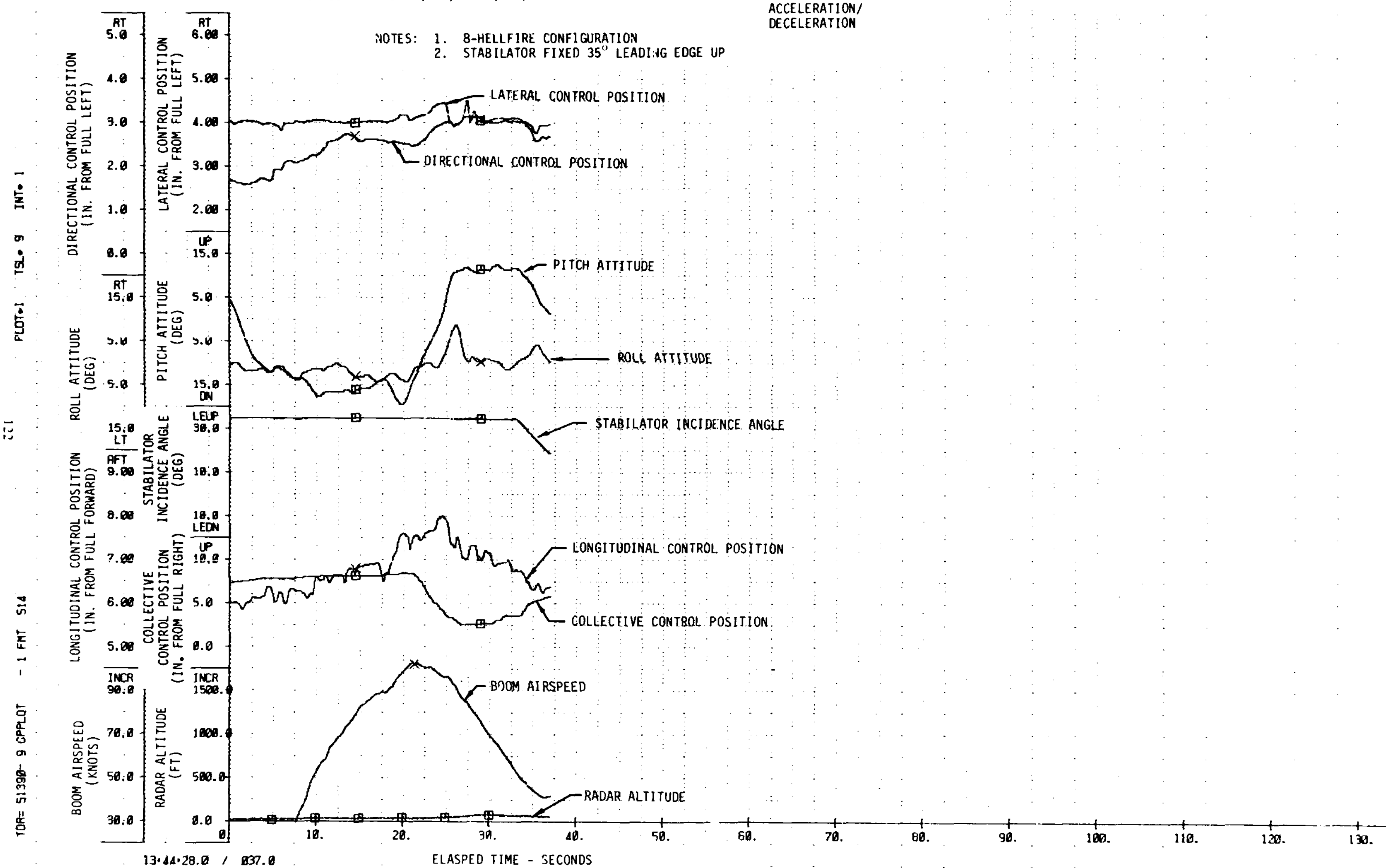
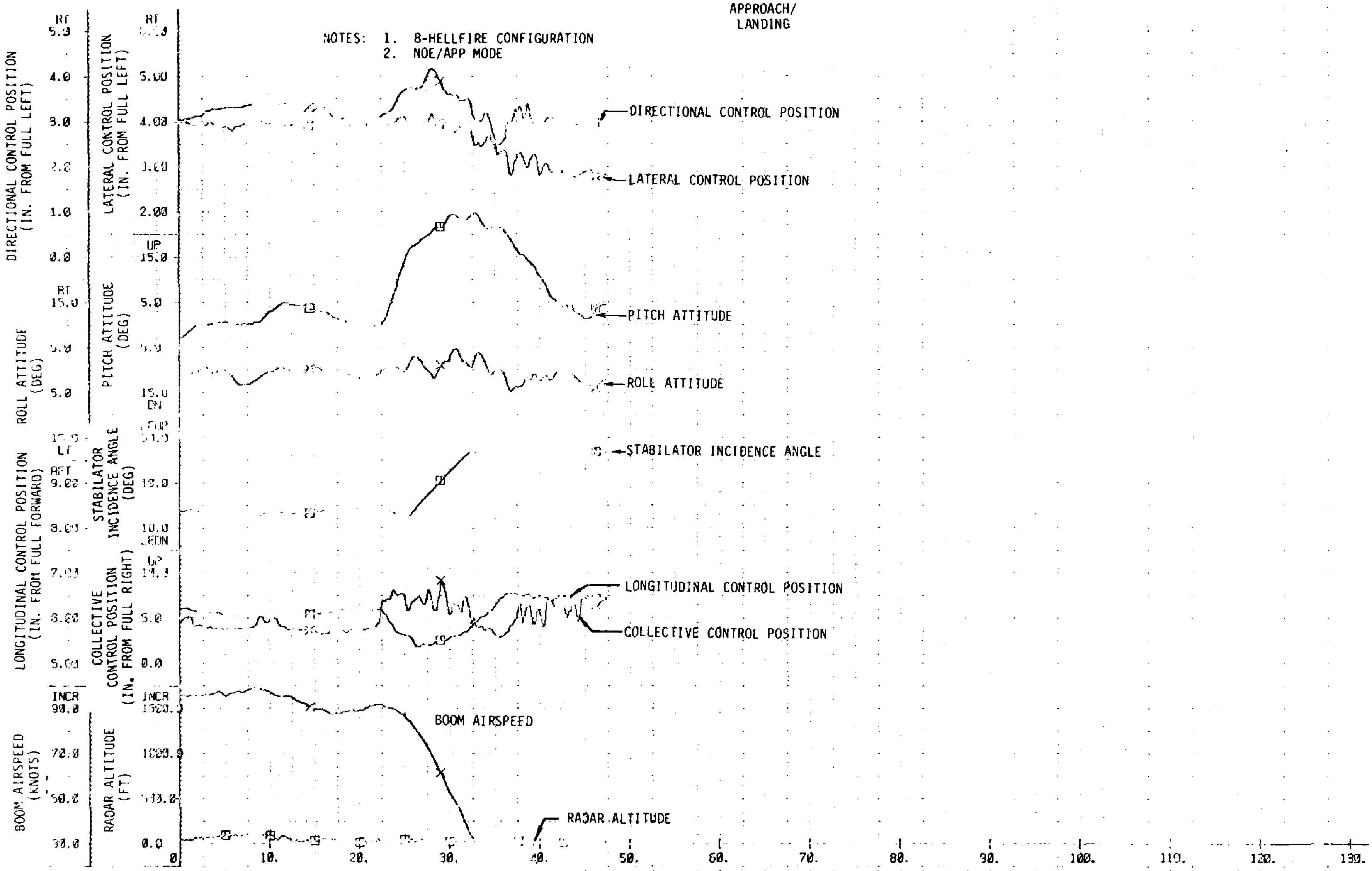


FIGURE 39
STABILATOR EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
14760	202.4 (FWD)	0.0 (MID)	660	16.5	289	CONTOUR APPROACH/ LANDING	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. NOE/APP MODE



TOP= 51365-13 GPPLOT - 1 FMT 514 PLOT=1 TSL= 13 INT= 1

FIGURE 40
YAW HARDOVER
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
14960	202.1 (FWD)	0.0 (MID)	4000	25.0	290	LEVEL	ON

NOTE: 8-HELLFIRE CONFIGURATION

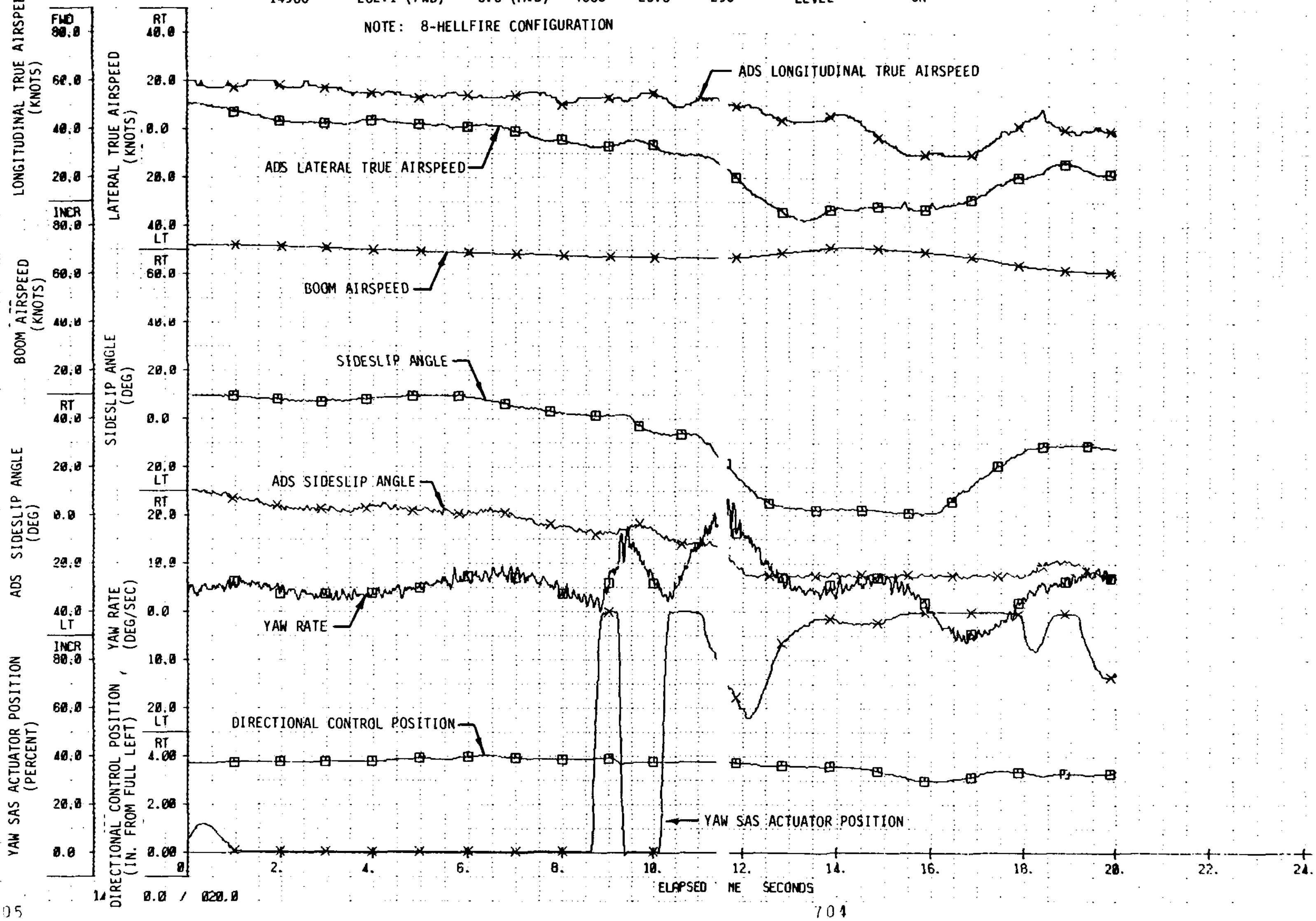
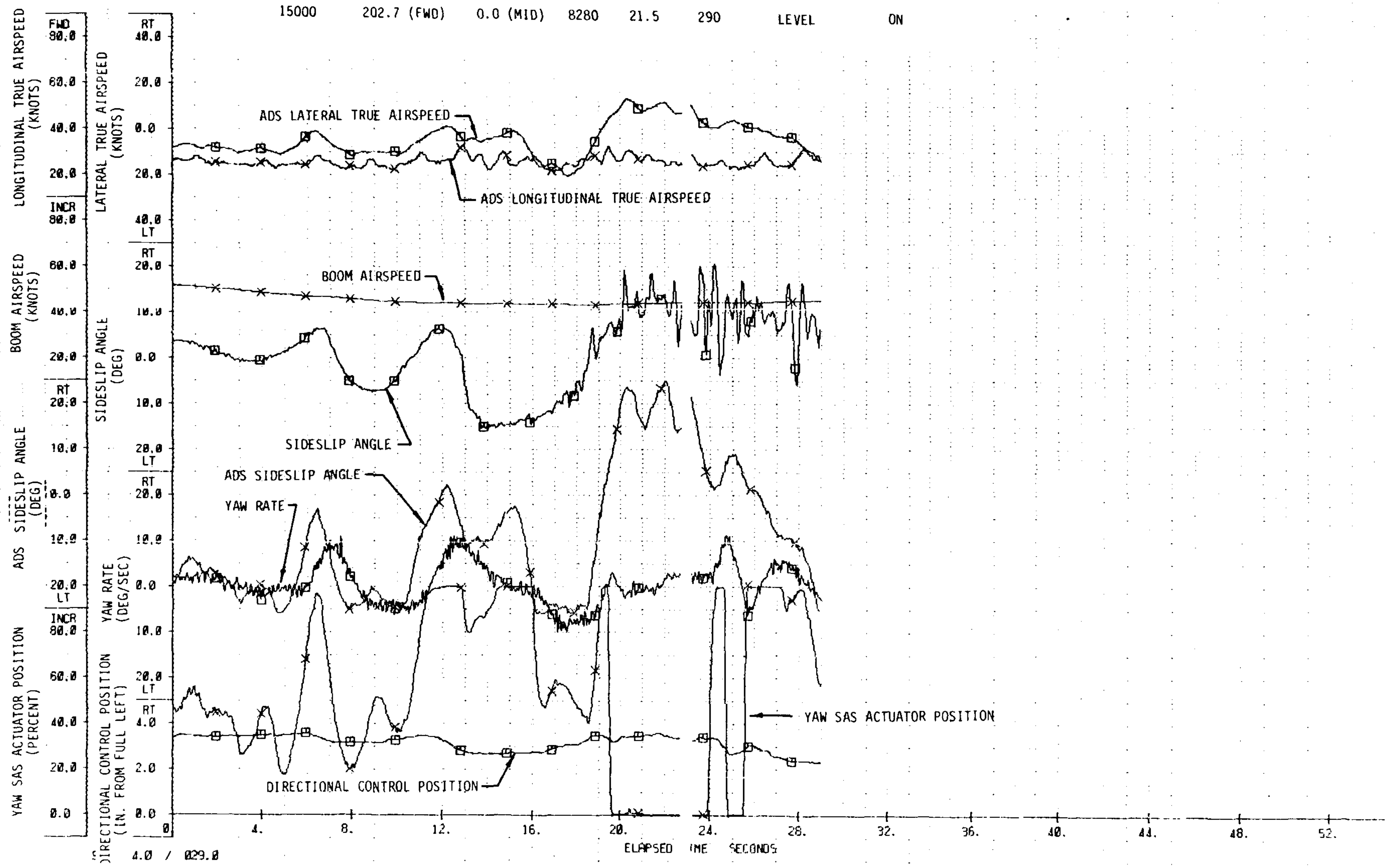


FIGURE 41
YAW HARDOVER
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15000	202.7 (FWD)	0.0 (MID)	8280	21.5	290	LEVEL	ON



TDR= 51656- 1 GPPLOT - 1 PLOT+1 TSL+ 3 INT+ 1

FIGURE 42
YAW HARDOVER
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15100	202.8 (FWD)	0.0 (MID)	1600	21.5	290	LEVEL	ON

NOTE: 8-HELLFIRE CONFIGURATION

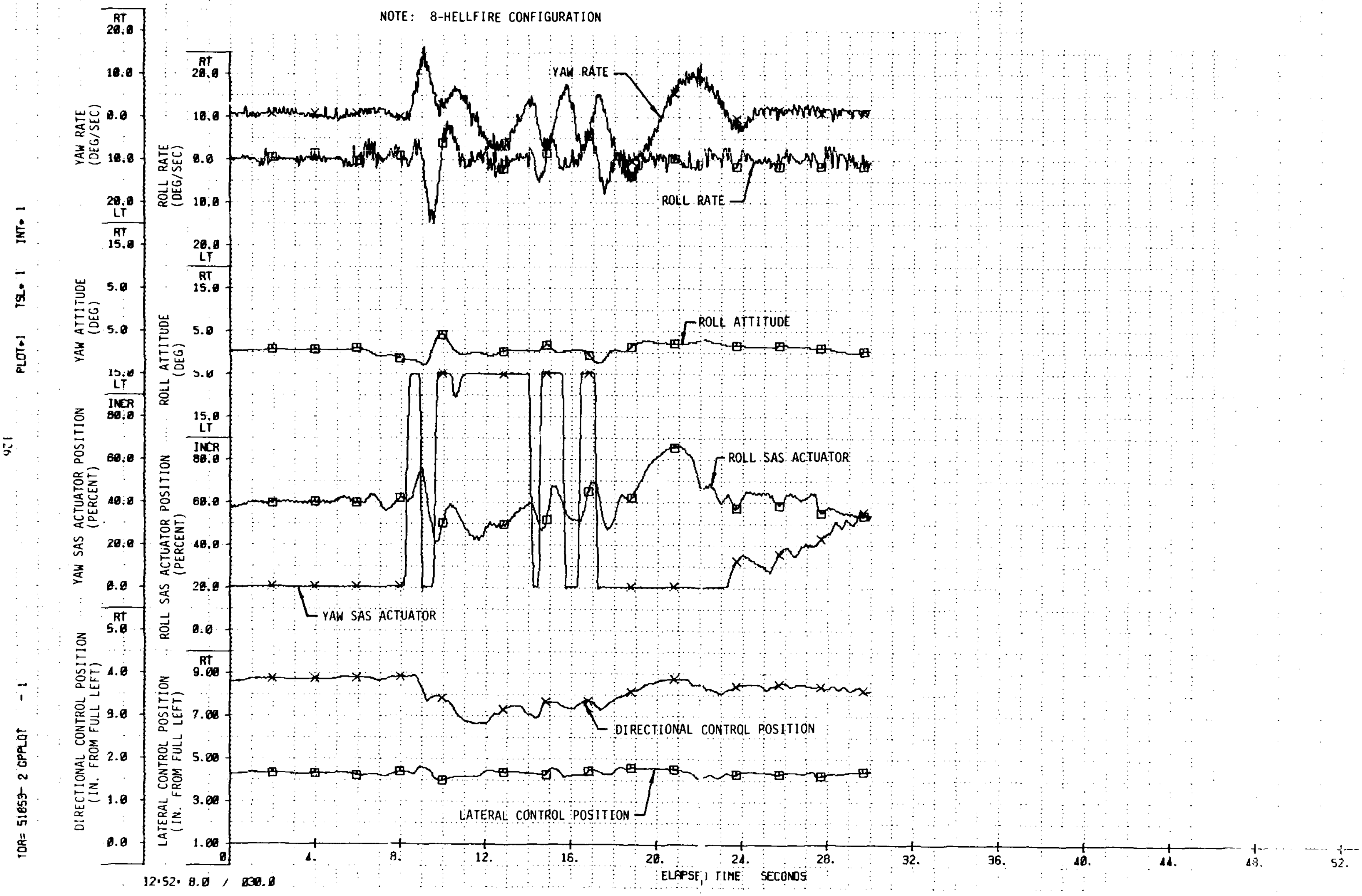
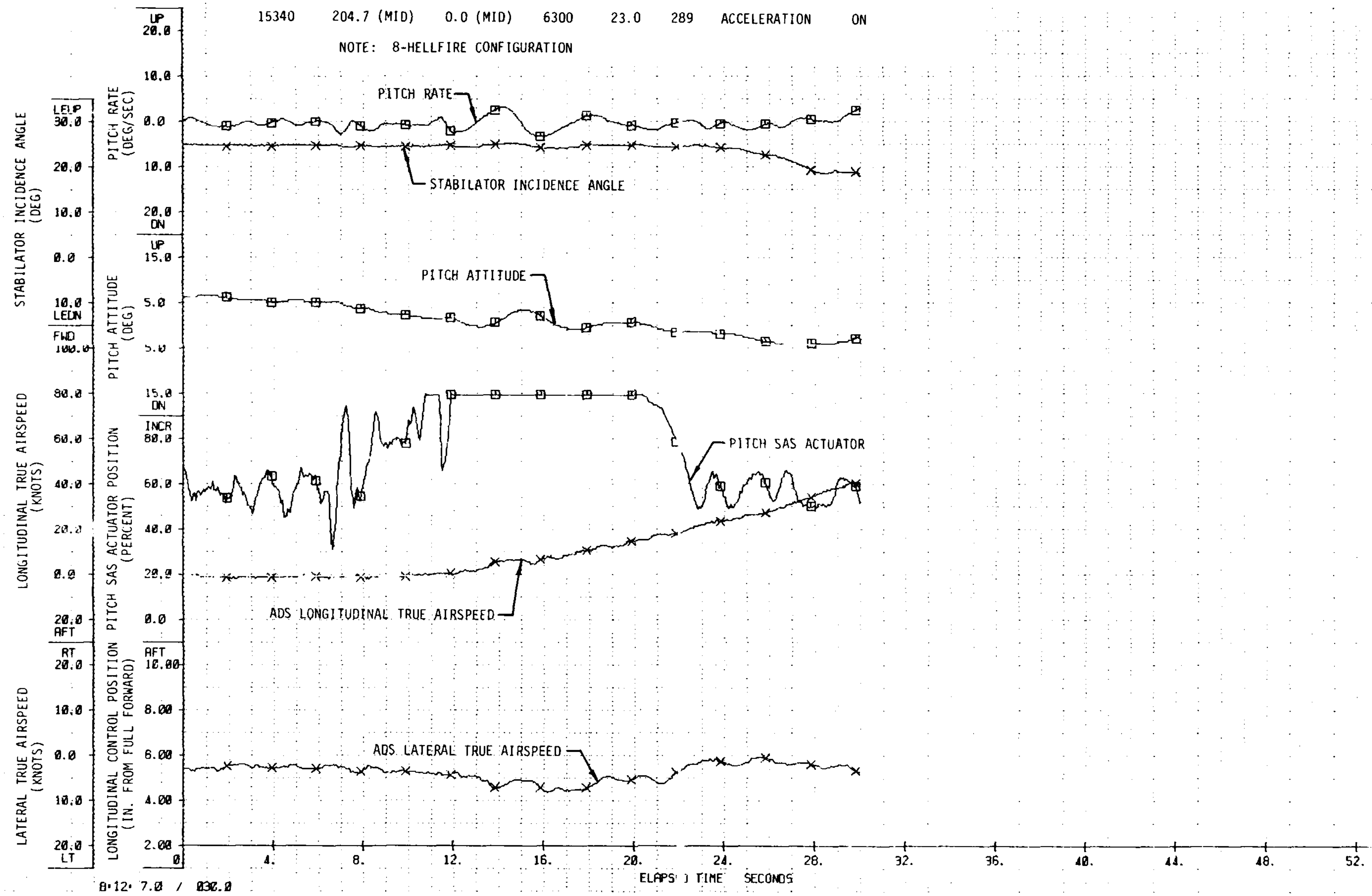


FIGURE 43
DASE EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (° C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15340	204.7 (MID)	0.0 (MID)	6300	23.0	289	ACCELERATION	ON

NOTE: 8-HELLFIRE CONFIGURATION

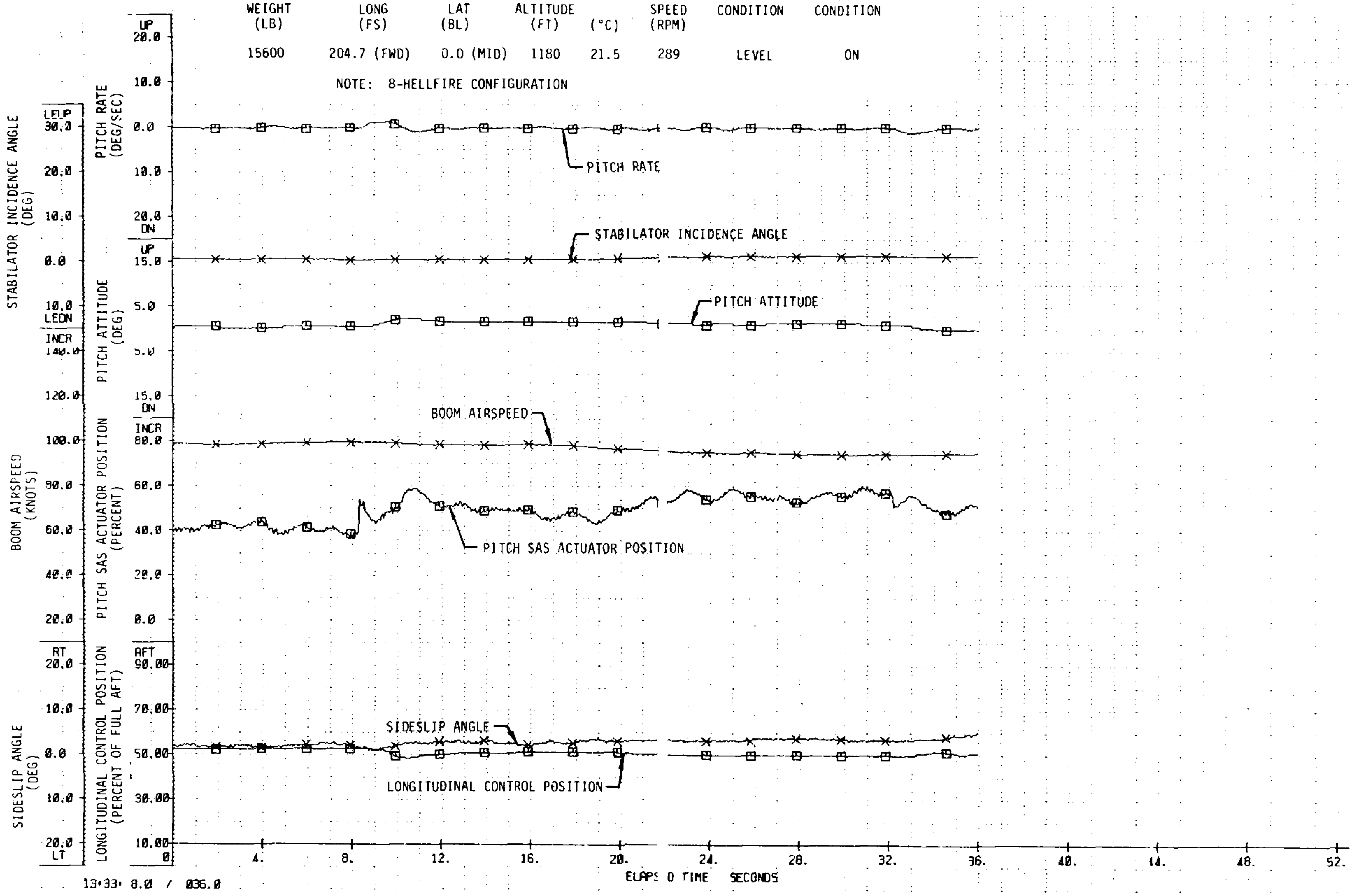


TDR= 51778- 4 GPPLOT - 1 PLOT-1 TSL- 4 INT- 1

FIGURE 44
DASE EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15600	204.7 (FWD)	0.0 (MID)	1180	21.5	289	LEVEL	ON

NOTE: 8-HELLFIRE CONFIGURATION



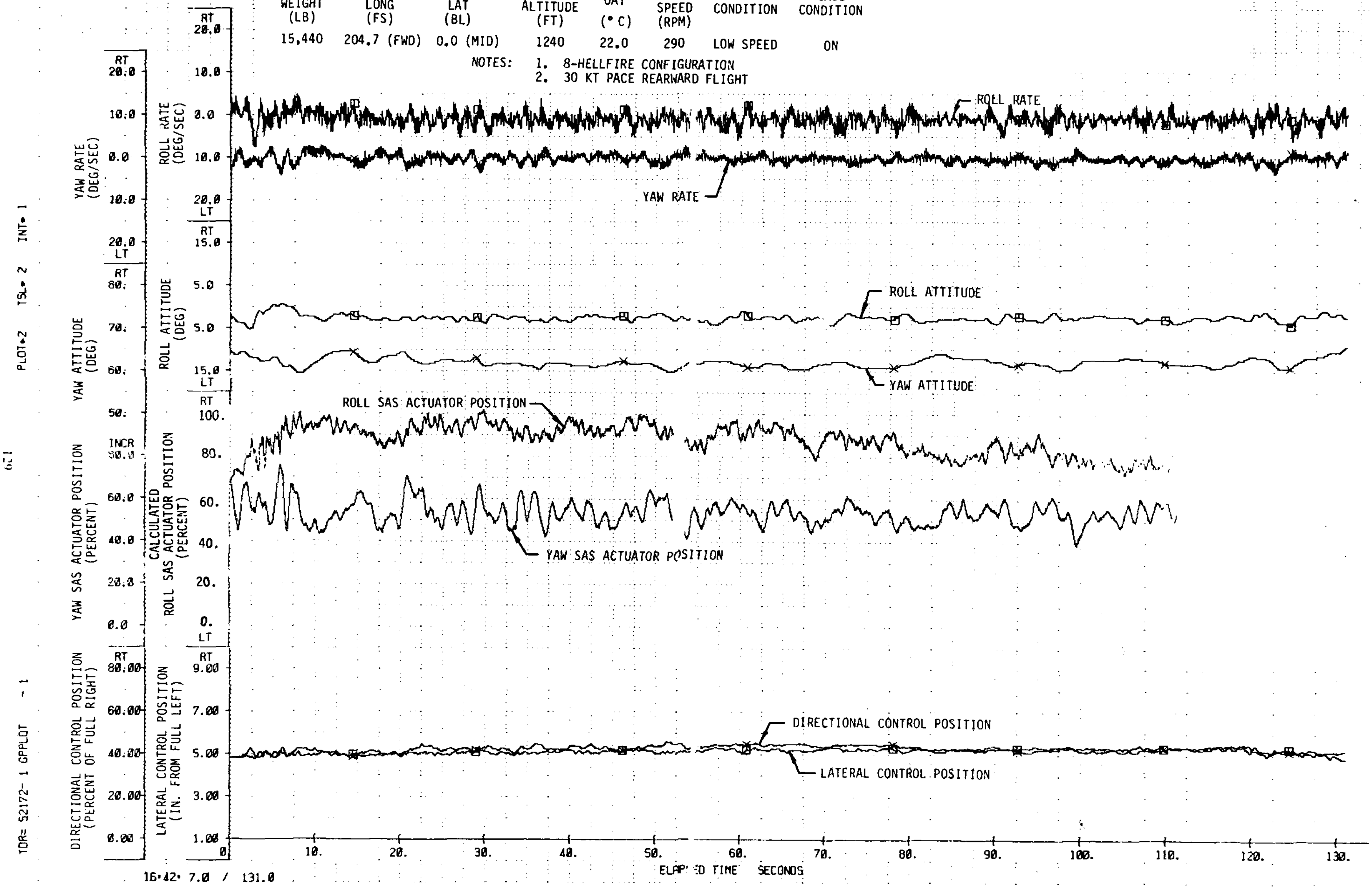
TOR= 51781- 5 GPPLOT - 1

PLOT+1 TSL+ 8 INT+ 1

FIGURE 45
DASE EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15,440	204.7 (FWD)	0.0 (MID)	1240	22.0	290	LOW SPEED	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. 30 KT PACE REARWARD FLIGHT



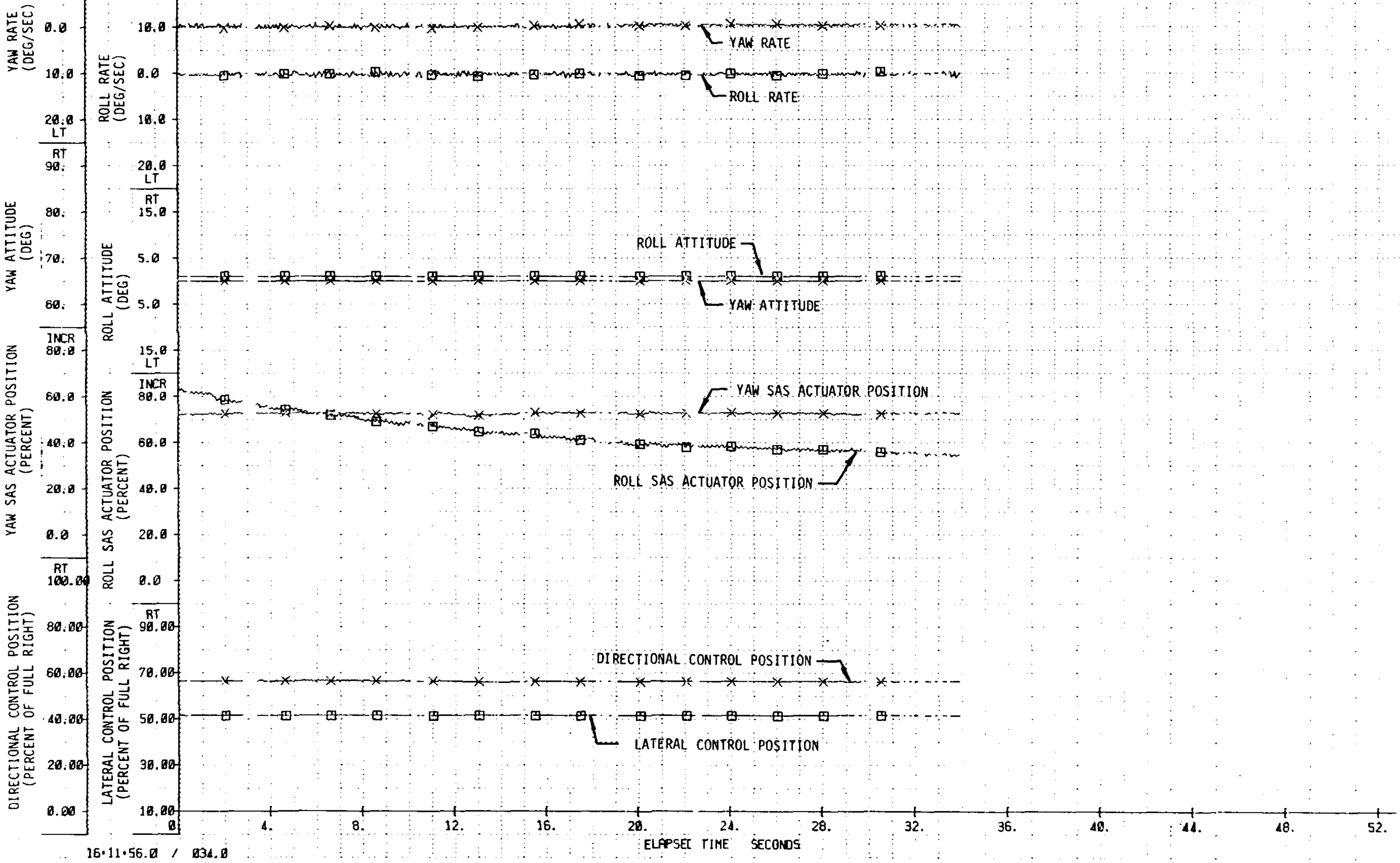
16•42• 7.0 / 131.0

TDR= 52172- 1 GPPLOT - 1 PLOT•2 TSL• 2 INT• 1

FIGURE 46
DASE EVALUATION
YAH-64 USA S/N 77-23258

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
15480	204.7 (FWD)	0.0 (MID)	1220	22.0	290	ON GROUND	ON

NOTE: 3-HELLFIRE CONFIGURATION



TOR= 51821- 1 GPPLOT - 1 PLOT• 1 TSL• 5 INT• 1

FIGURE 47
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15180	202.5 (FWD)	0.0 (MID)	720	16.5	289	LOW SPEED

NOTES:
1. PILOT SEAT
2. STABILATOR FIXED 25° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

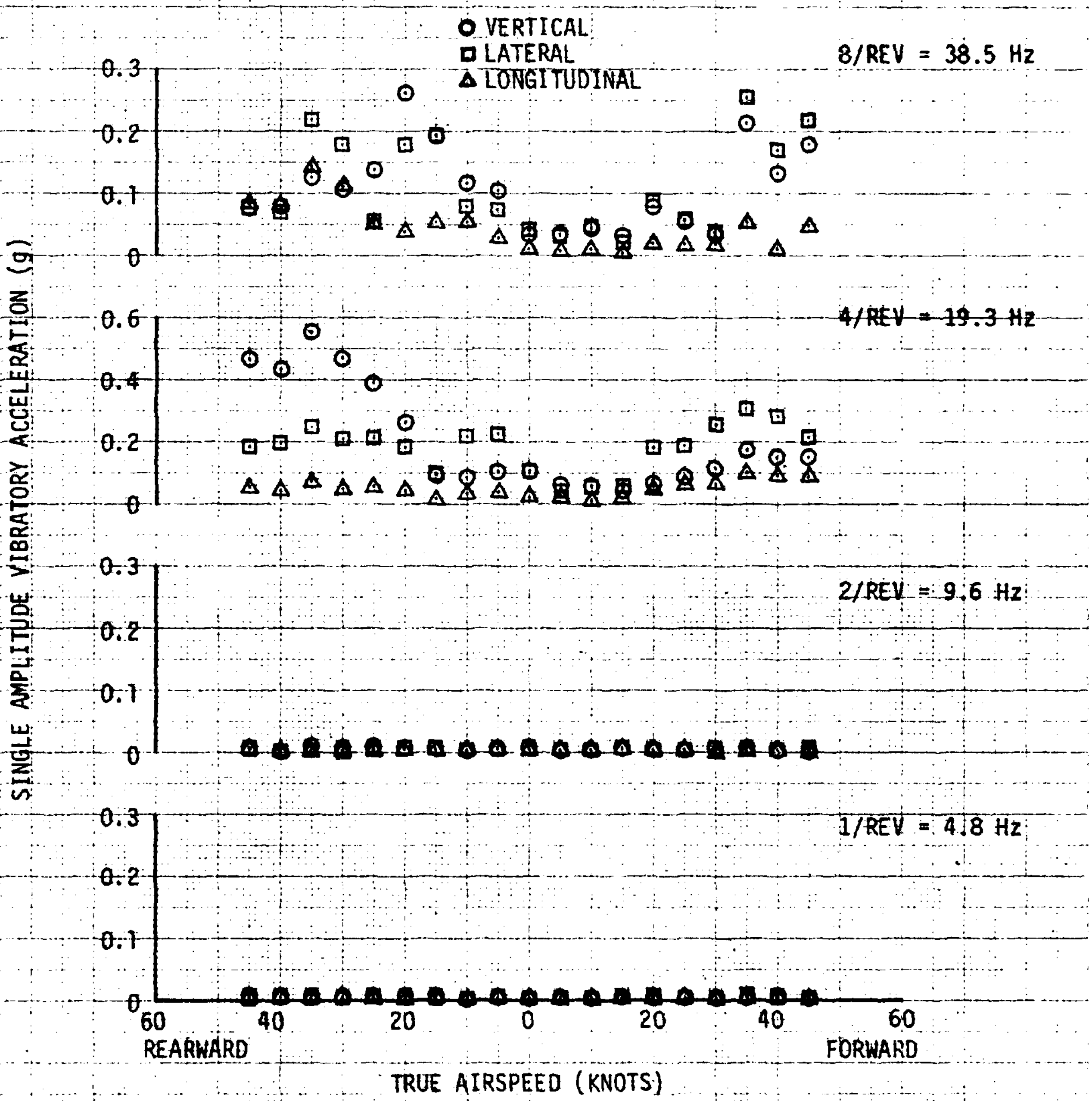


FIGURE 48
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15180	202.5 (FWD)	0.0 (MID)	720	16.5	289	LOW SPEED

NOTES:
1. PILOT FLOOR
2. STABILATOR FIXED 25° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

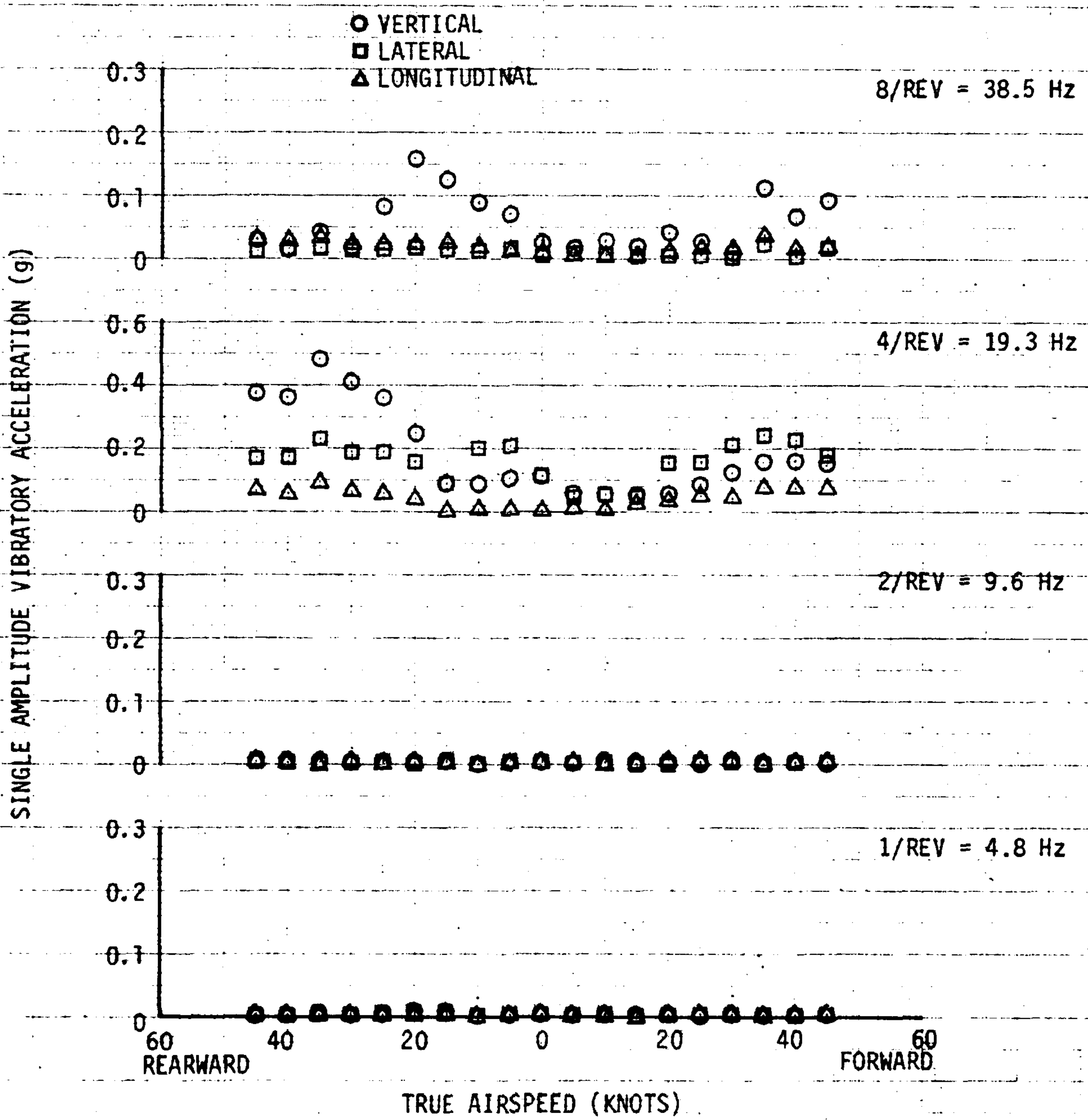


FIGURE 49
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15180	202.5 (FWD)	0.0 (MID)	720	16.5	289	LOW SPEED

NOTES:
1. COPILOT SEAT
2. STABILATOR FIXED 25° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

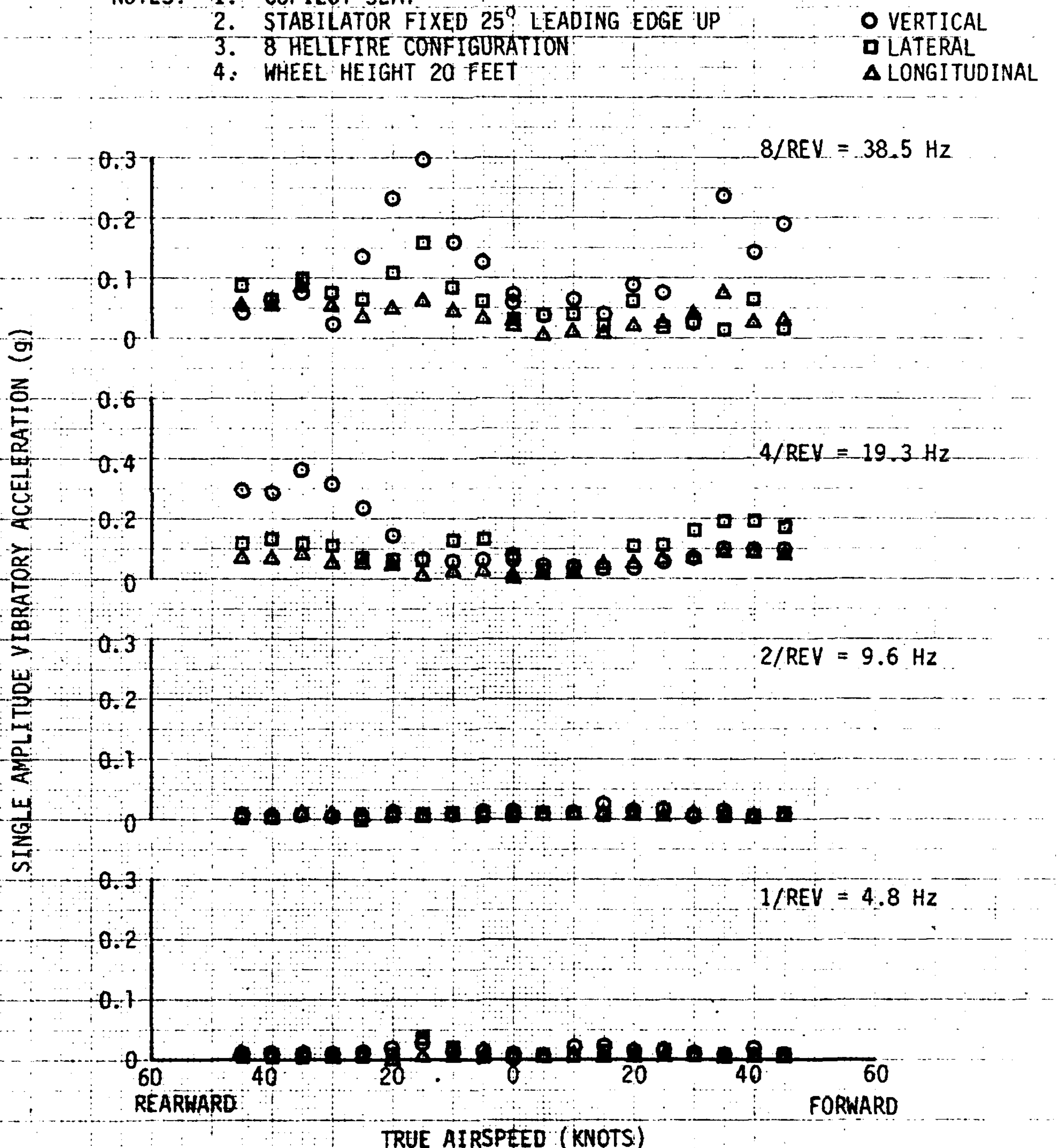


FIGURE 50
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15180	202.5 (FWD)	0.0 (MID)	720	16.5	289	LOW SPEED

NOTES:
1. COPILOT FLOOR
2. STABILATOR FIXED 25° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

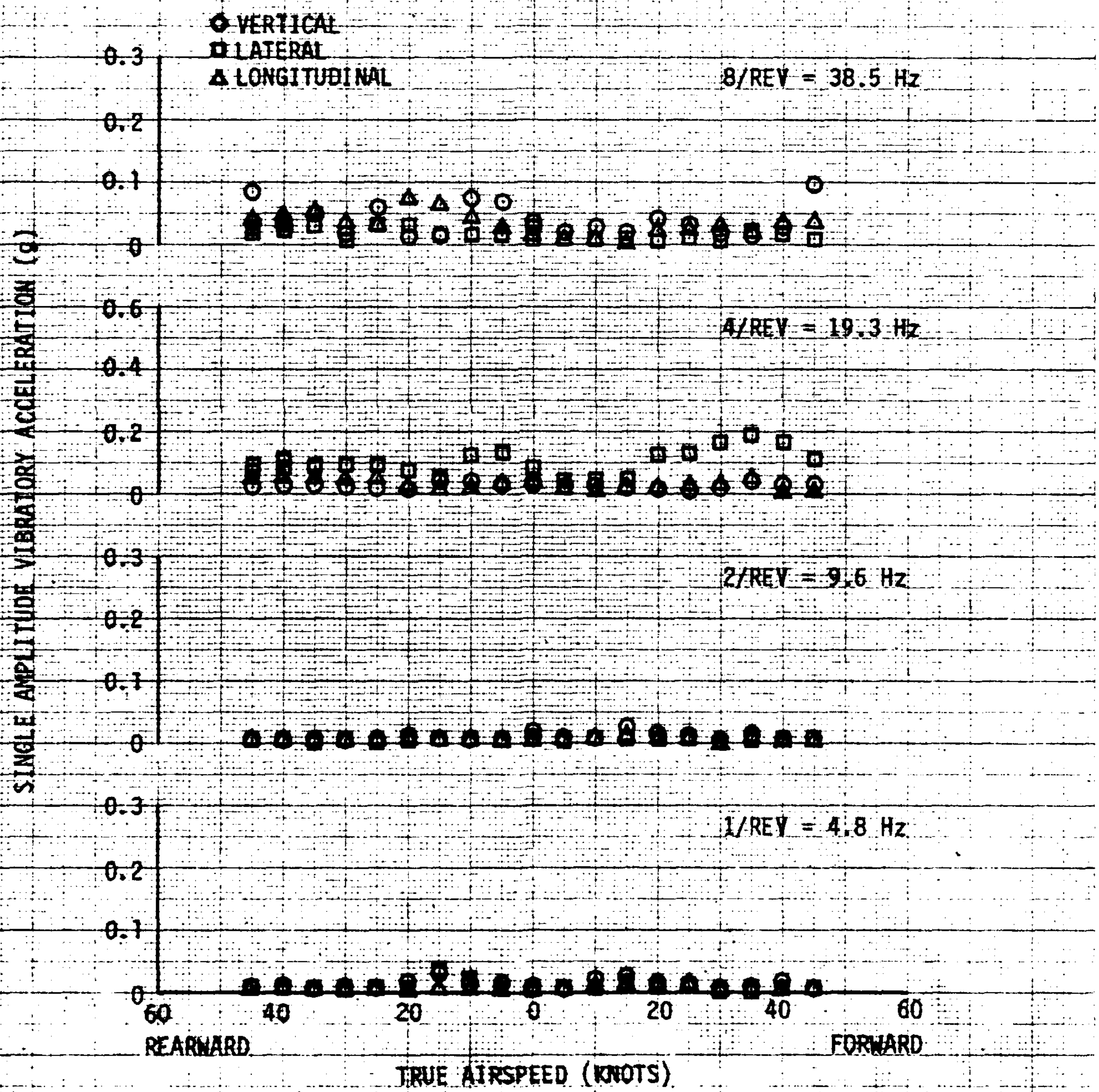


FIGURE [illegible]

VIBRATION CHARACTERISTICS

YAH-64 USA S/N 77-23258

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
15180	202.5 (FWD)	0.0 (MID)	[illegible]	16.5	289	LOW SPEED

NOTES:
1. AIRCRAFT CG
2. STABILATOR FIXED 25 LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

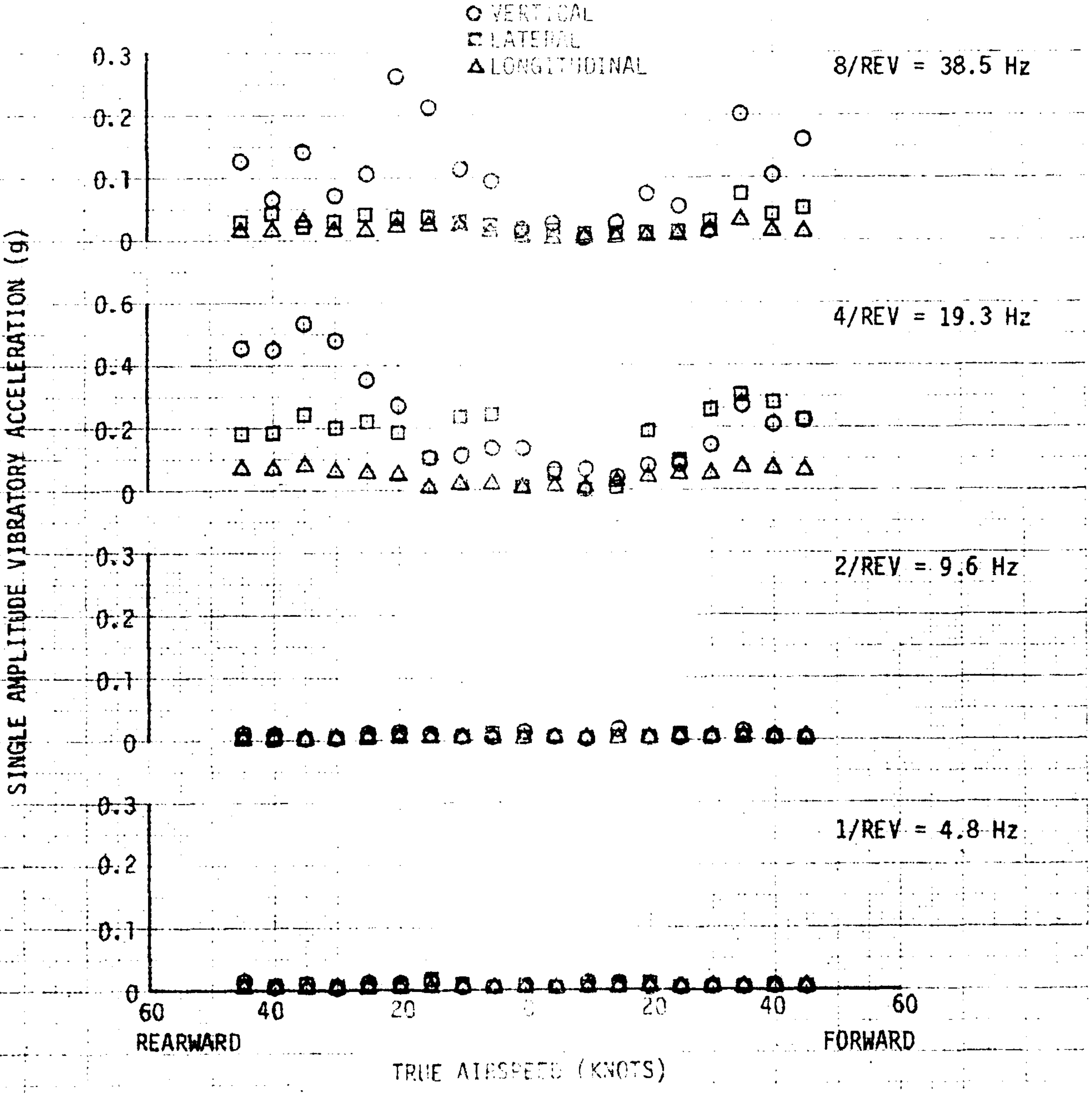


FIGURE 52
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14720	202.5 (FWD)	0.0 (MID)	700	16.5	289	LOW SPEED

NOTES:
1. PILOT SEAT
2. STABILATOR FIXED 35° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

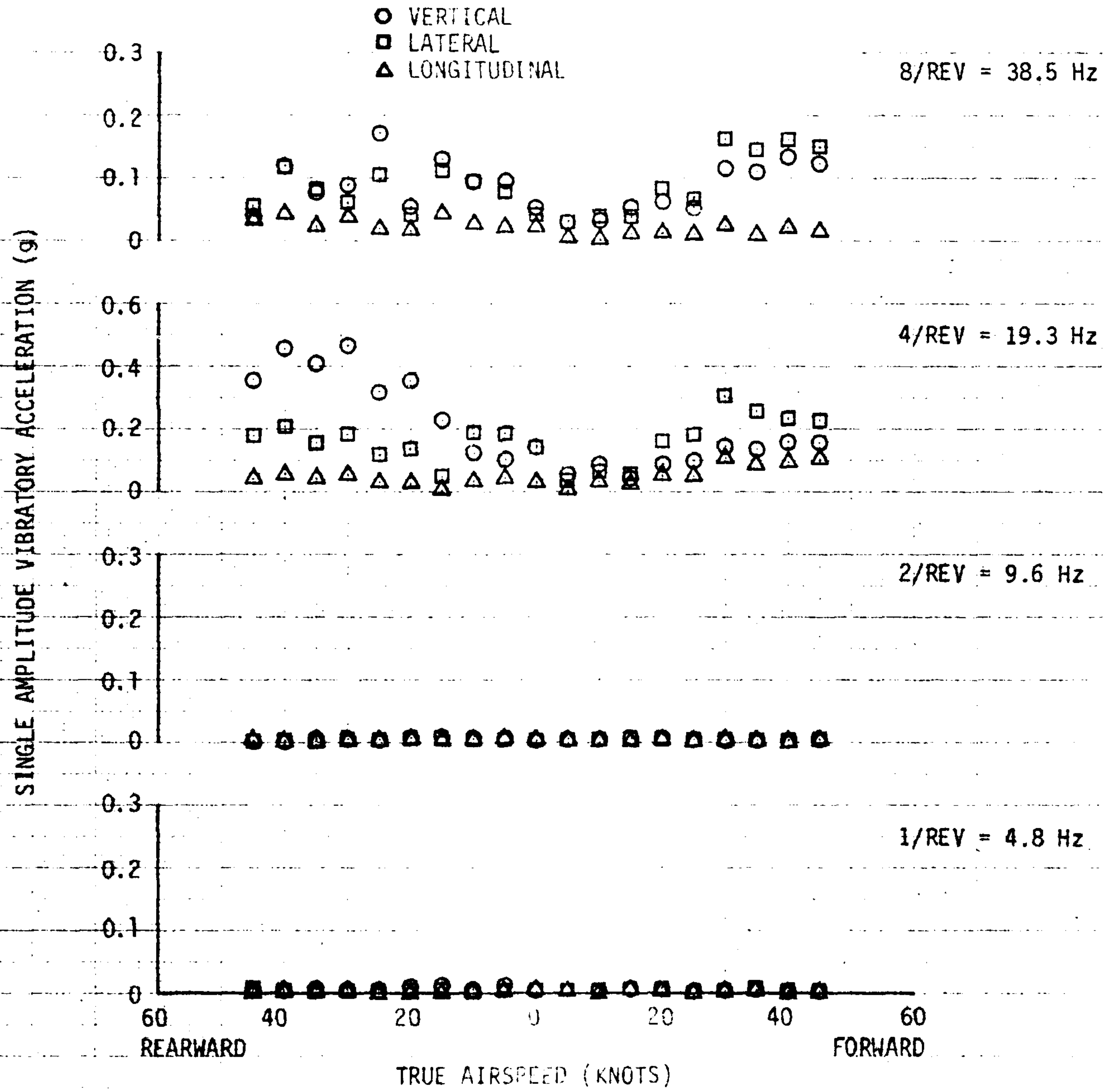


FIGURE 53
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14720	202.5 (FWD)	0.0 (MID)	700	16.5	289	LOW SPEED

NOTES: 1. PILOT FLOOR
2. STABILATOR FIXED 35° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

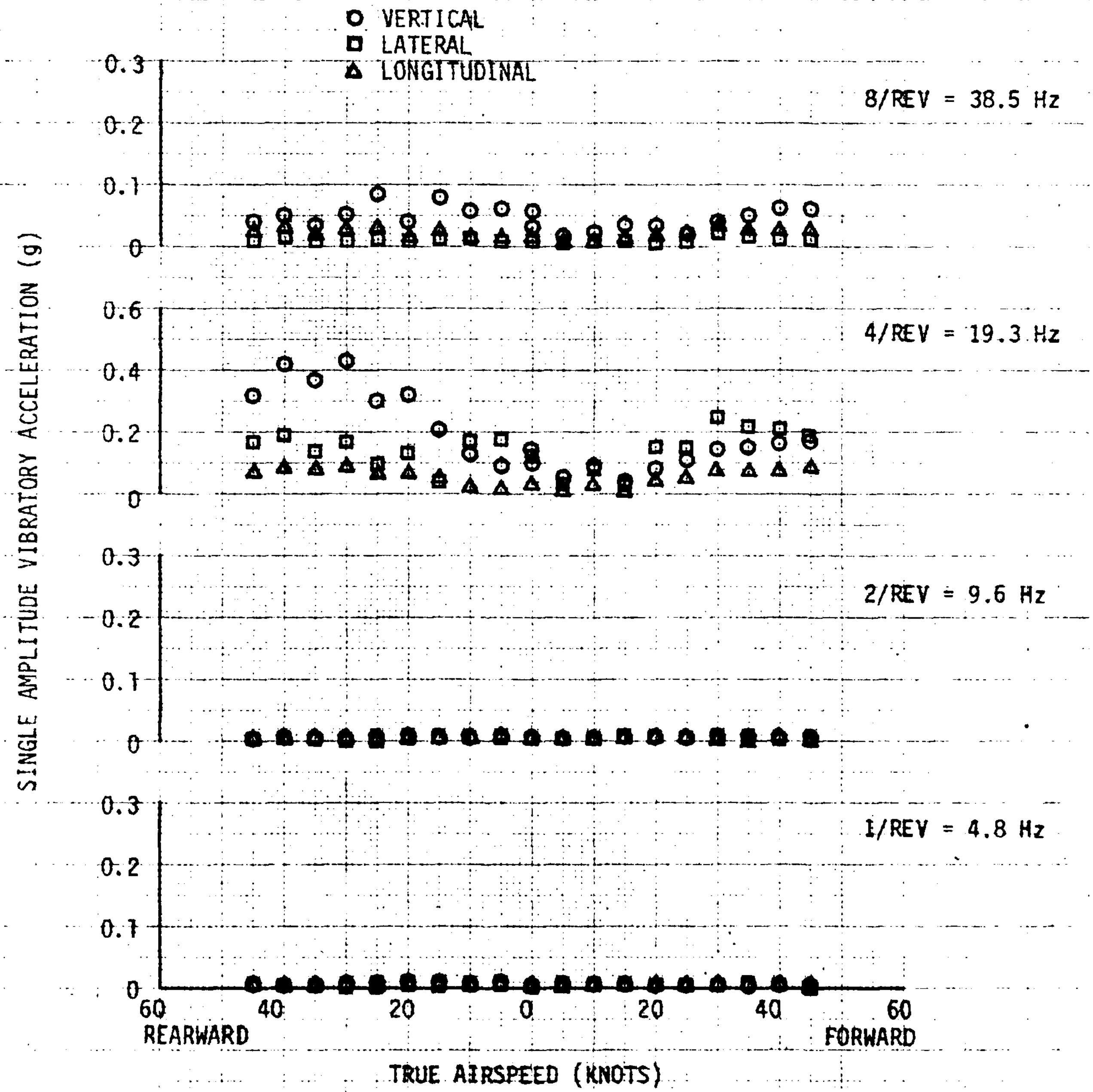


FIGURE 54
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14720	202.5 (FWD)	0.0 (MID)	700	16.5	289	LOW SPEED

NOTES:
1. COPILOT SEAT
2. STABILATOR FIXED 35° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

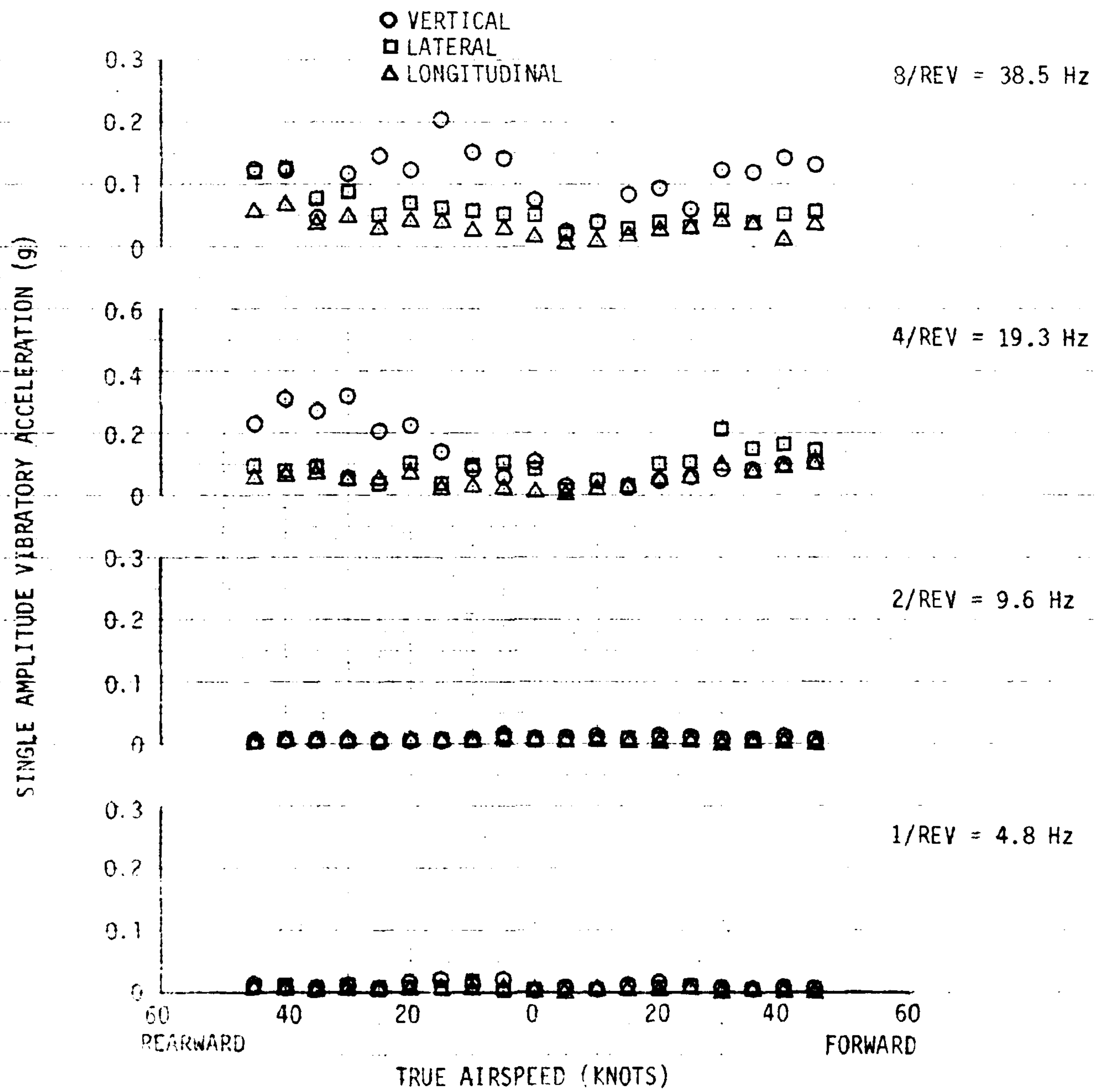


FIGURE 55
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14720	202.5 (FWD)	0.0 (MID)	700	16.5	289	LOW SPEED

NOTES: 1. COPILOT FLOOR
2. STABILATOR FIXED 35° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

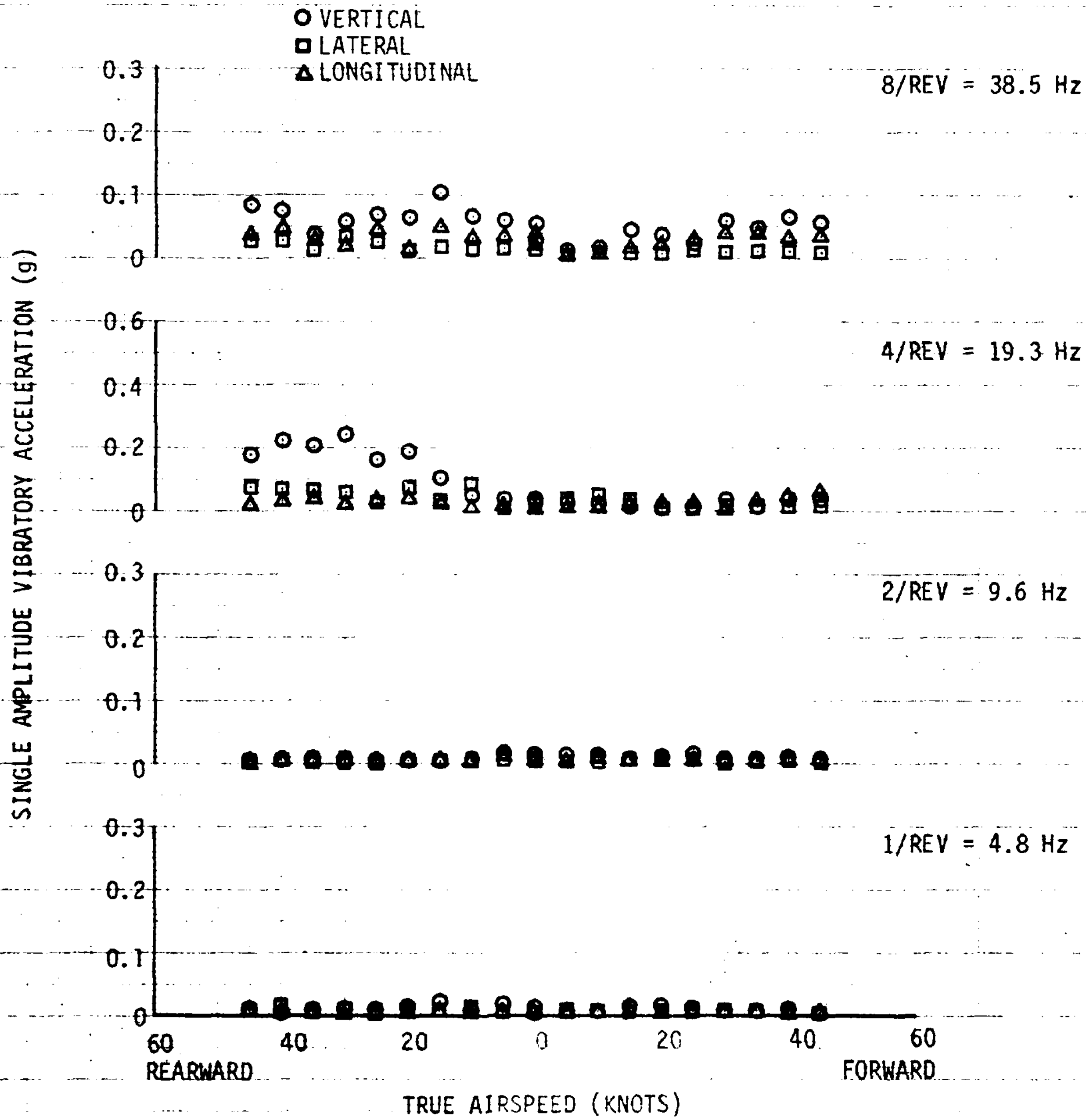


FIGURE 56
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14720	202.5 (FWD)	0.0 (MID)	700	16.5	289	LOW SPEED

NOTES:
1. AIRCRAFT CG
2. STABILATOR FIXED 35° LEADING EDGE UP
3. 8 HELLFIRE CONFIGURATION
4. WHEEL HEIGHT 20 FEET

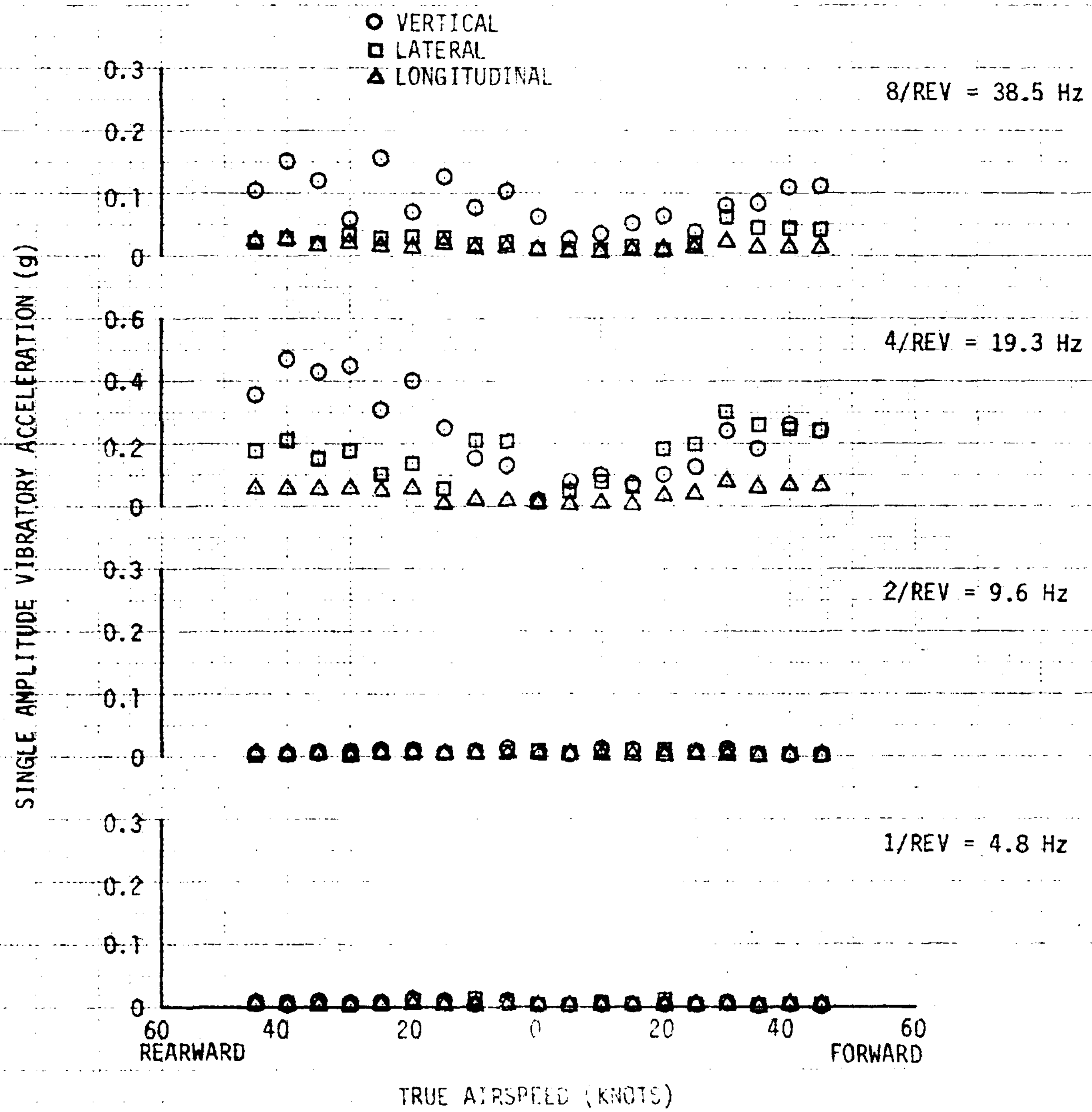


FIGURE 57
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15240	202.4 (FWD)	0.0 (MID)	6580	27.0	289	LOW SPEED

NOTES:
1. PILOT SEAT
2. 8 HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

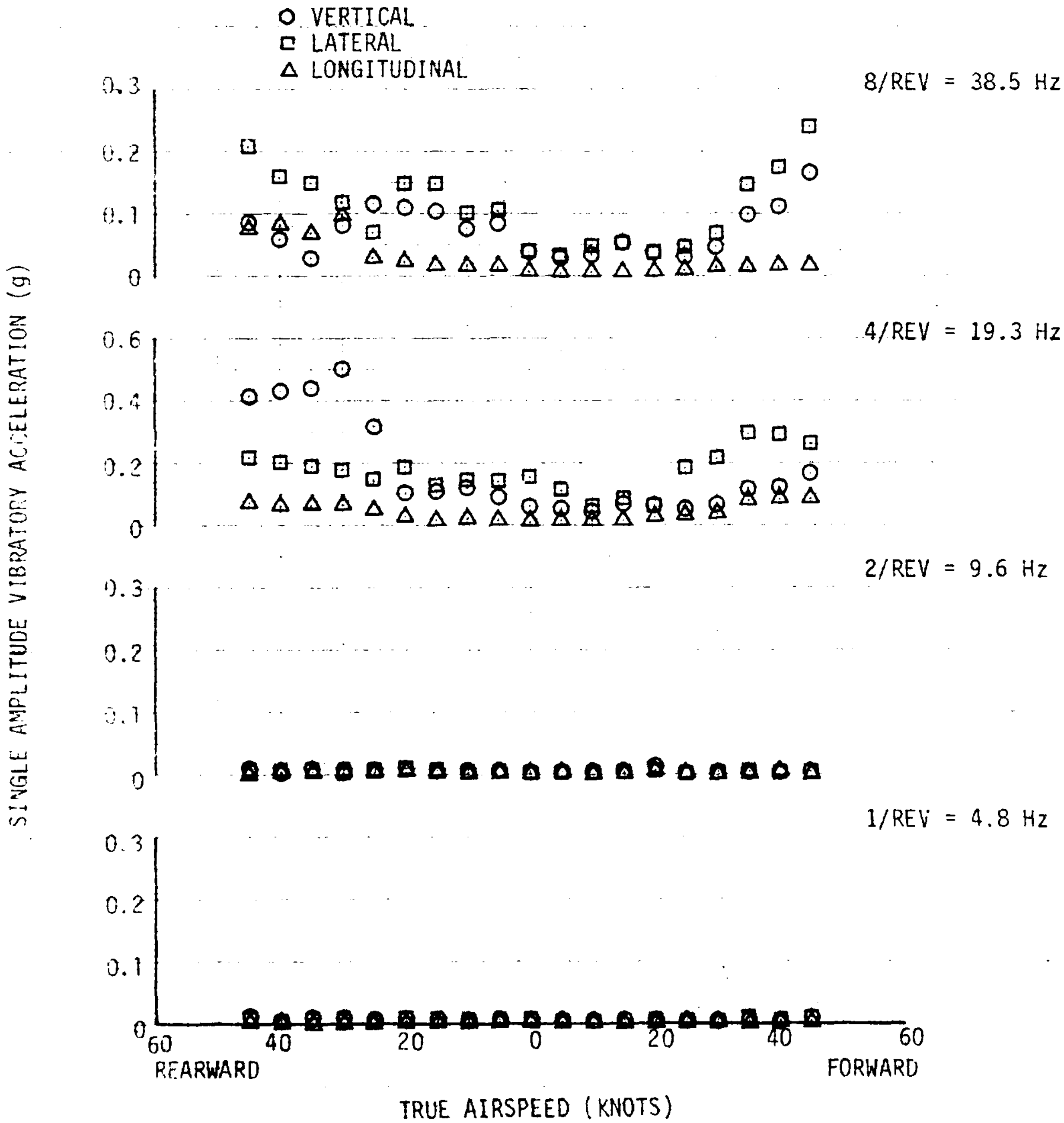


FIGURE 58
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15240	202.4 (FWD)	0.0 (MID)	6580	27.0	289	LOW SPEED

NOTES: 1. COPILOT SEAT
2. 8 HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

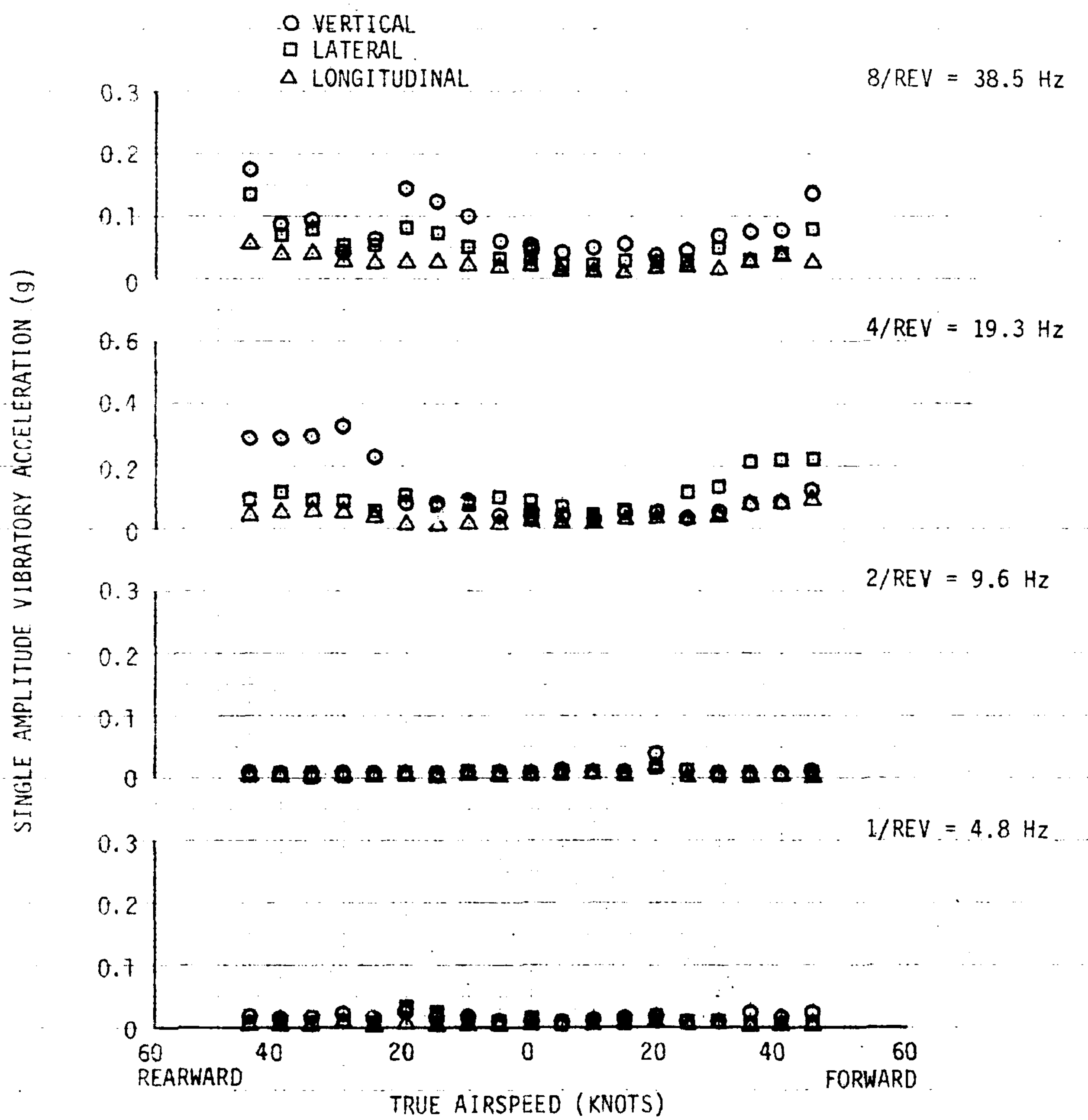


FIGURE 59

VIBRATION CHARACTERISTICS

YAH-64 USA S/N 77-23258

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
15240	202.5 (FWD)	0.0 (MID)	10540	4.5	289	LOW SPEED

NOTES:
1. PILOT SEAT
2. 8 HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

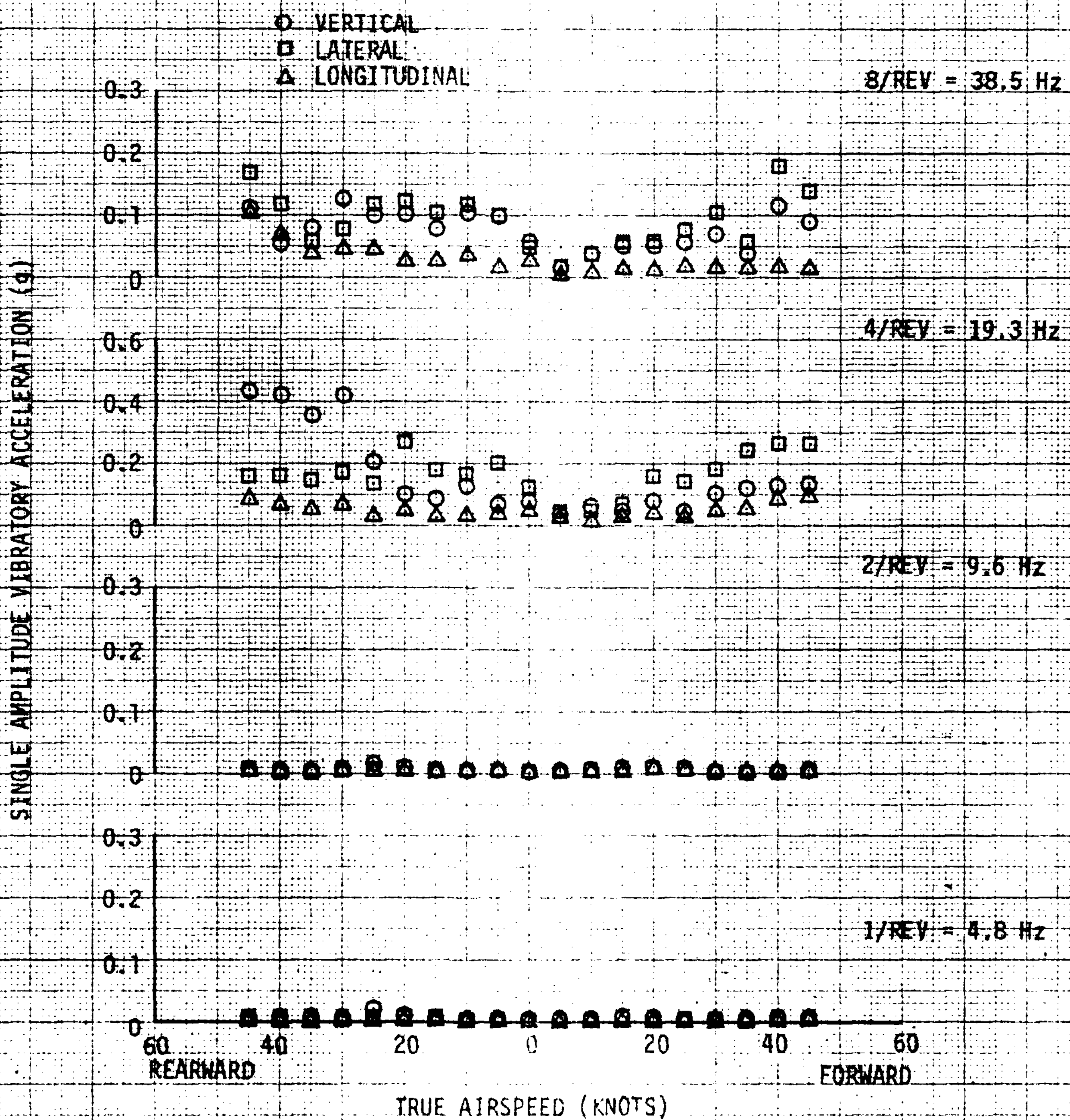


FIGURE 60
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15240	202.5 (FWD)	0.0 (MID)	10540	4.5	289	LOW SPEED

NOTES: 1. COPILOT SEAT
2. 8 HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

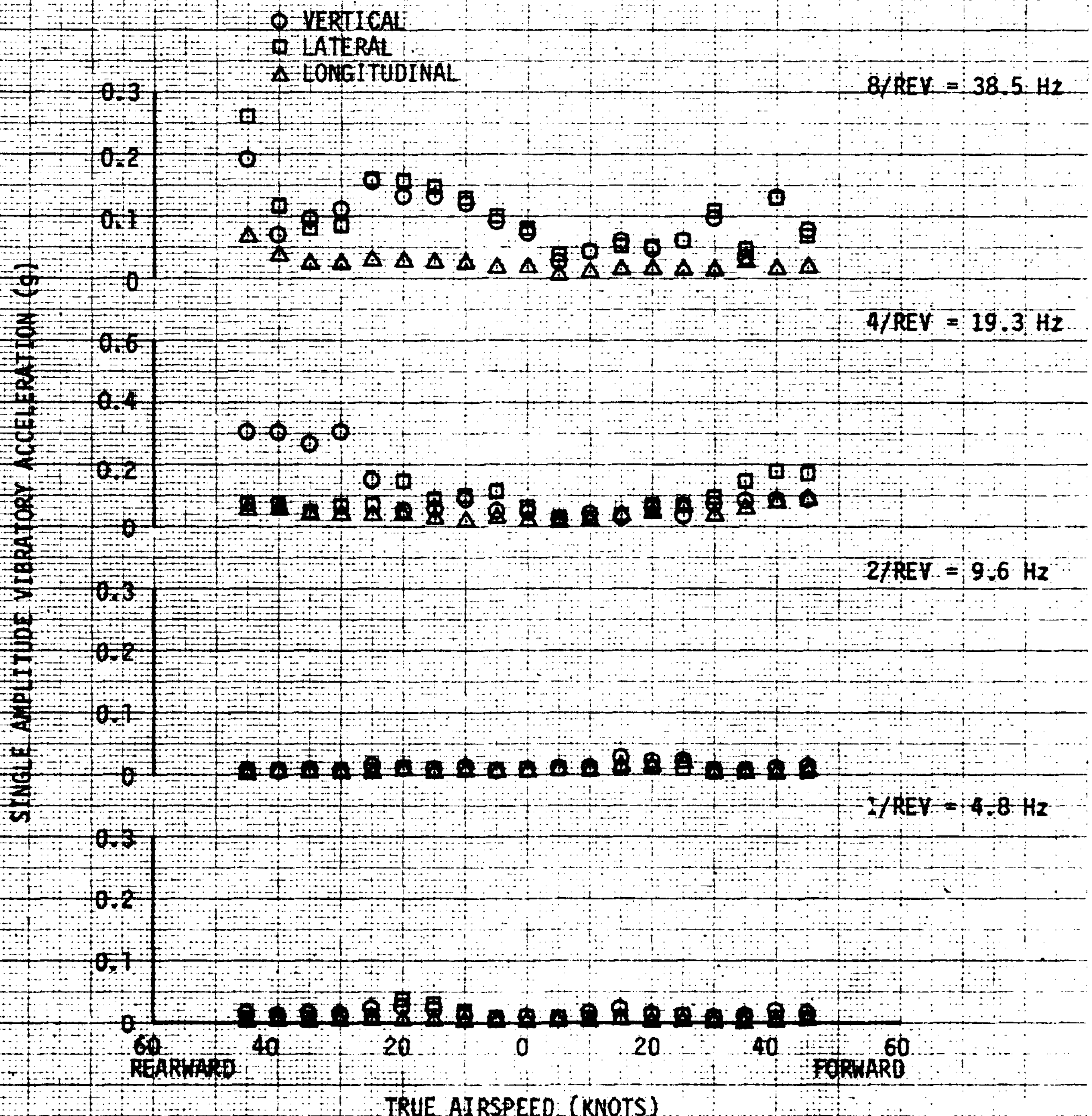


FIGURE 61
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15560	210.9 (FWD)	0.0 (MID)	5680	21.0	290	LOW SPEED

NOTES: 1. PILOT SEAT
2. 16-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

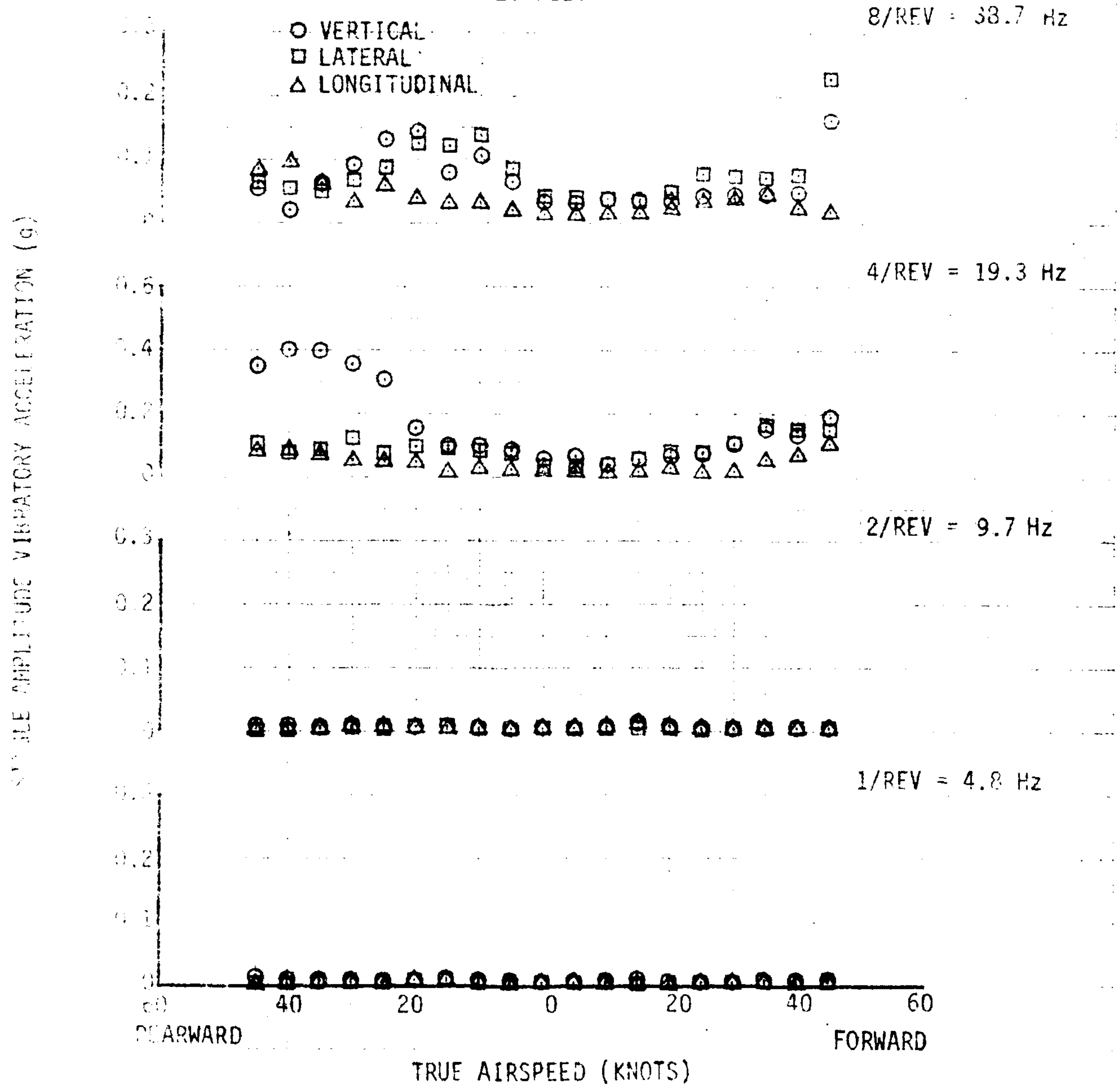
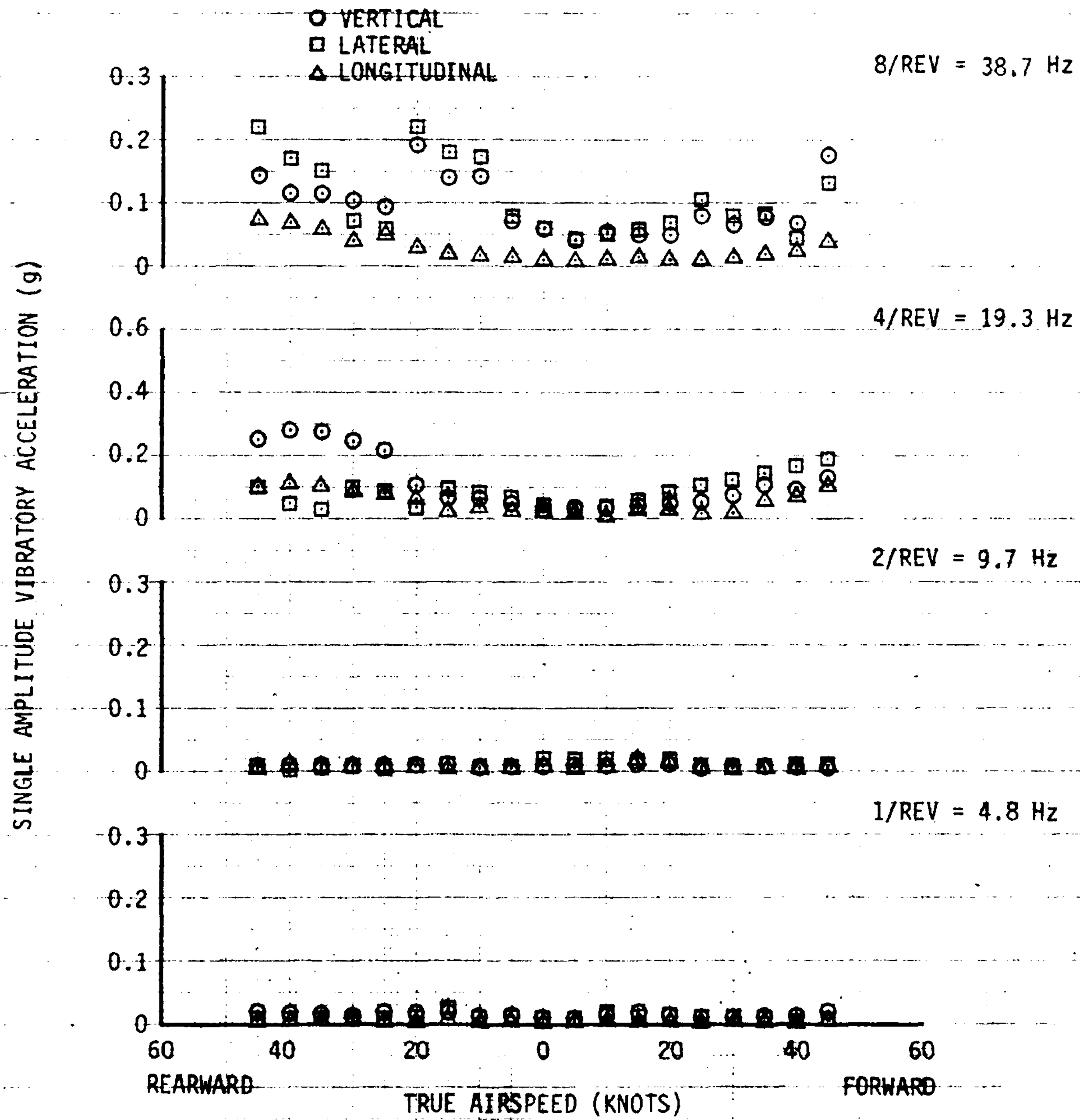


FIGURE 62
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
16560	201.9 (FWD)	0.0 (MID)	5680	21.0	290	LOW SPEED

NOTES: 1. COPILOT SEAT
2. 16-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET



VIBRATION [illegible]TICS
YAH-64 [illegible] 77-23258

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
14960	202.3 (FWD)	0.0 (MI[illegible]	[illegible]	27.5	289	LOW SPEED

NOTES: 1. PILOT SEAT
2. 8 HELLFIRE CON[illegible]
3. WHEEL HEIGHT [illegible]

○ VERTICAL
□ LATERAL
△ LONGITUDINAL

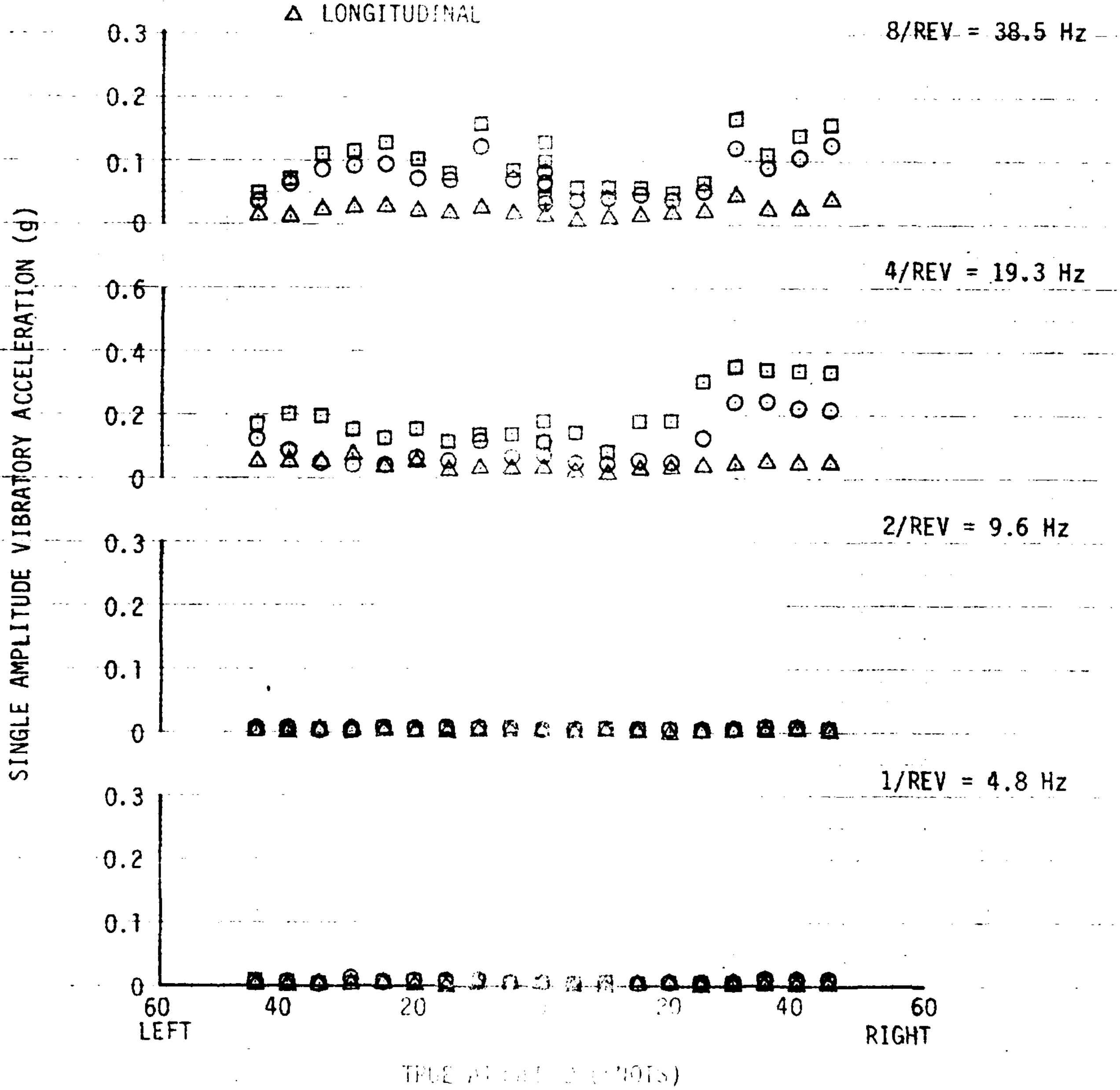


FIGURE 64
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14960	202.3 (FWD)	0.0 (MID)	6660	27.5	289	LOW SPEED

NOTES: 1. COPILOT SEAT
2. 8 HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

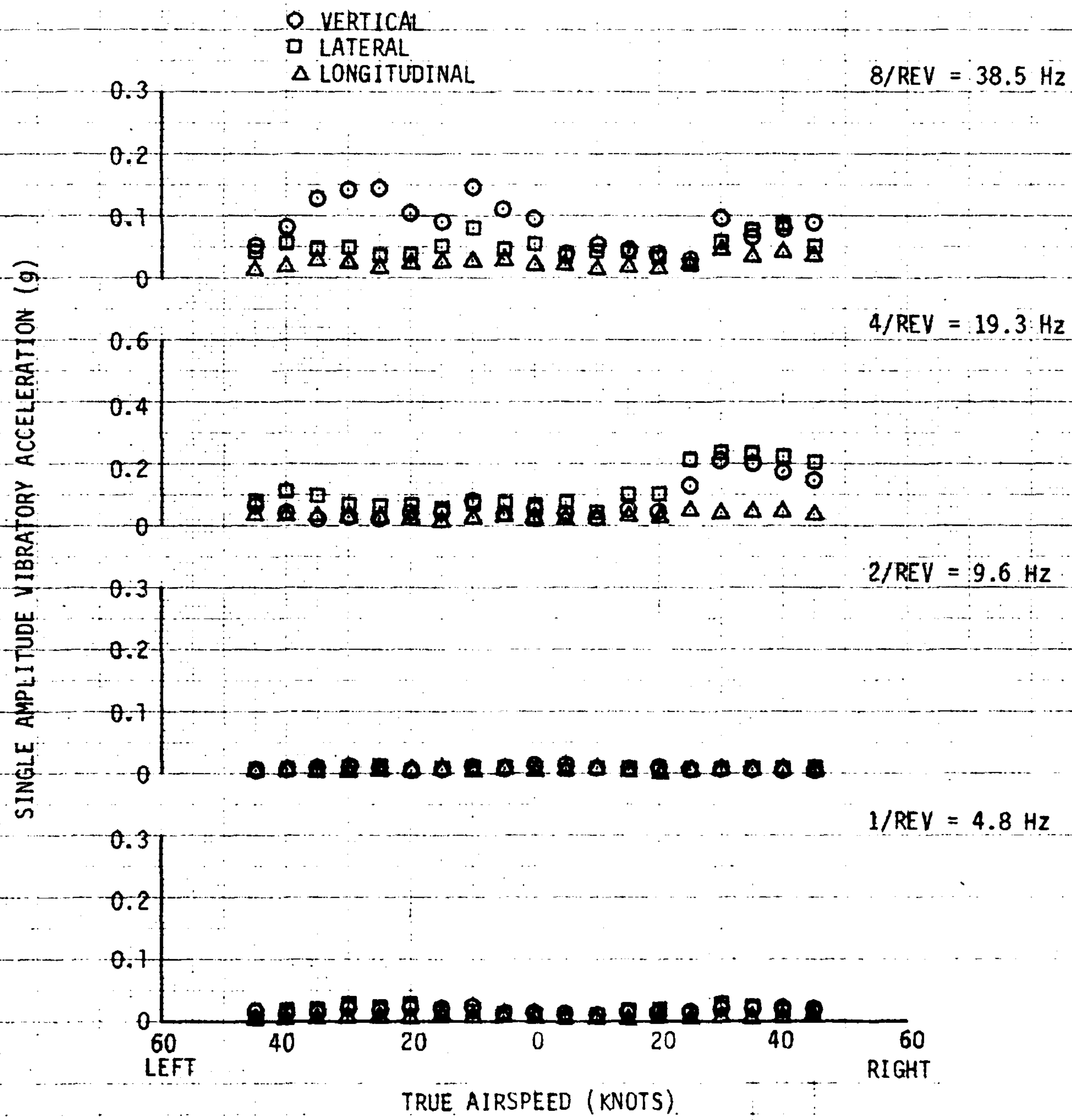


FIGURE 65
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14960	202.3 (FWD)	0.0 (MID)	6660	27.5	289	LOW SPEED

NOTES: 1. AIRCAFT CG
2. 8 HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

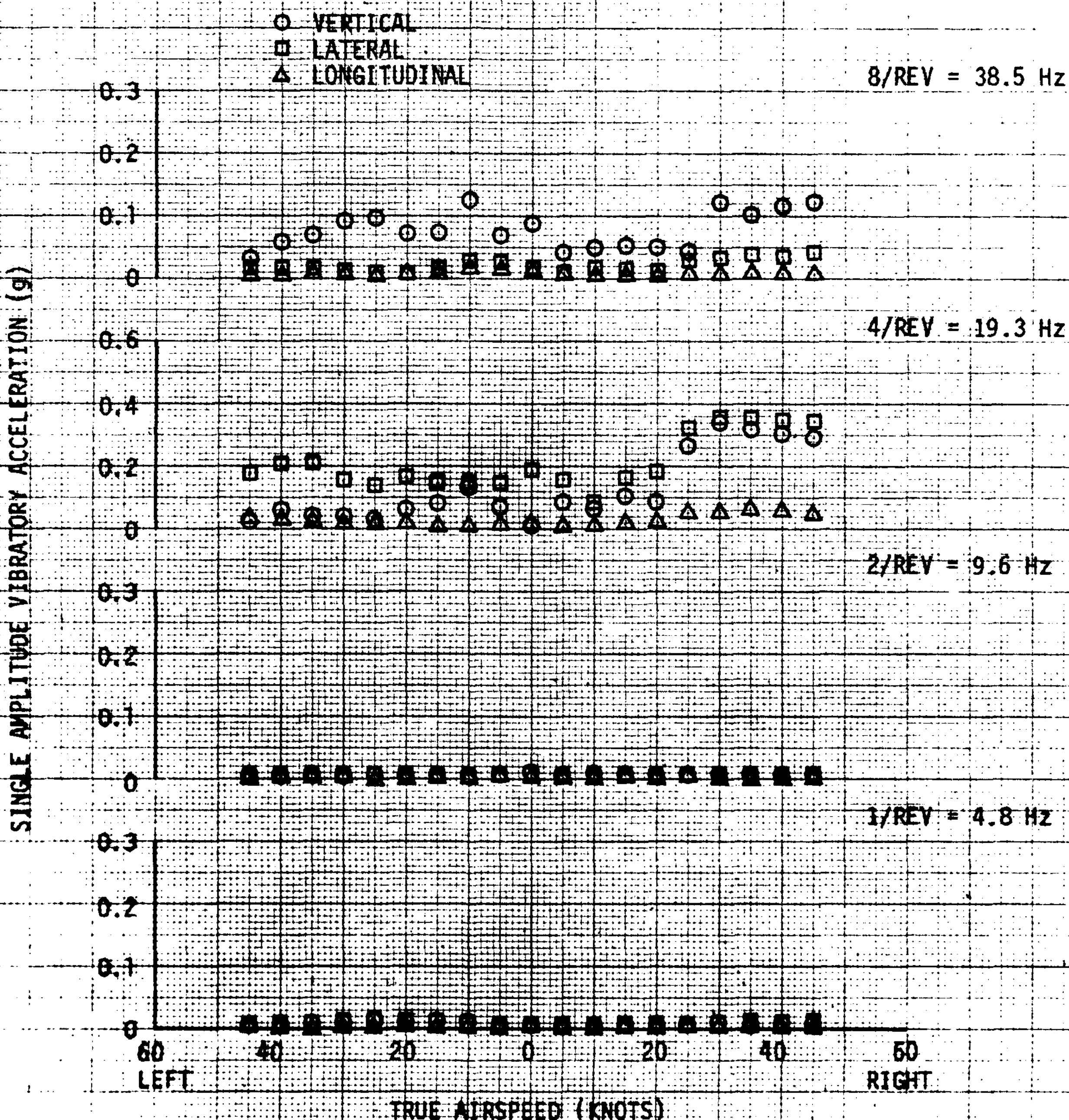


FIGURE 66
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14440	202.5 (FWD)	0.0 (MID)	10600	4.5	289	LOW SPEED

NOTES: 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

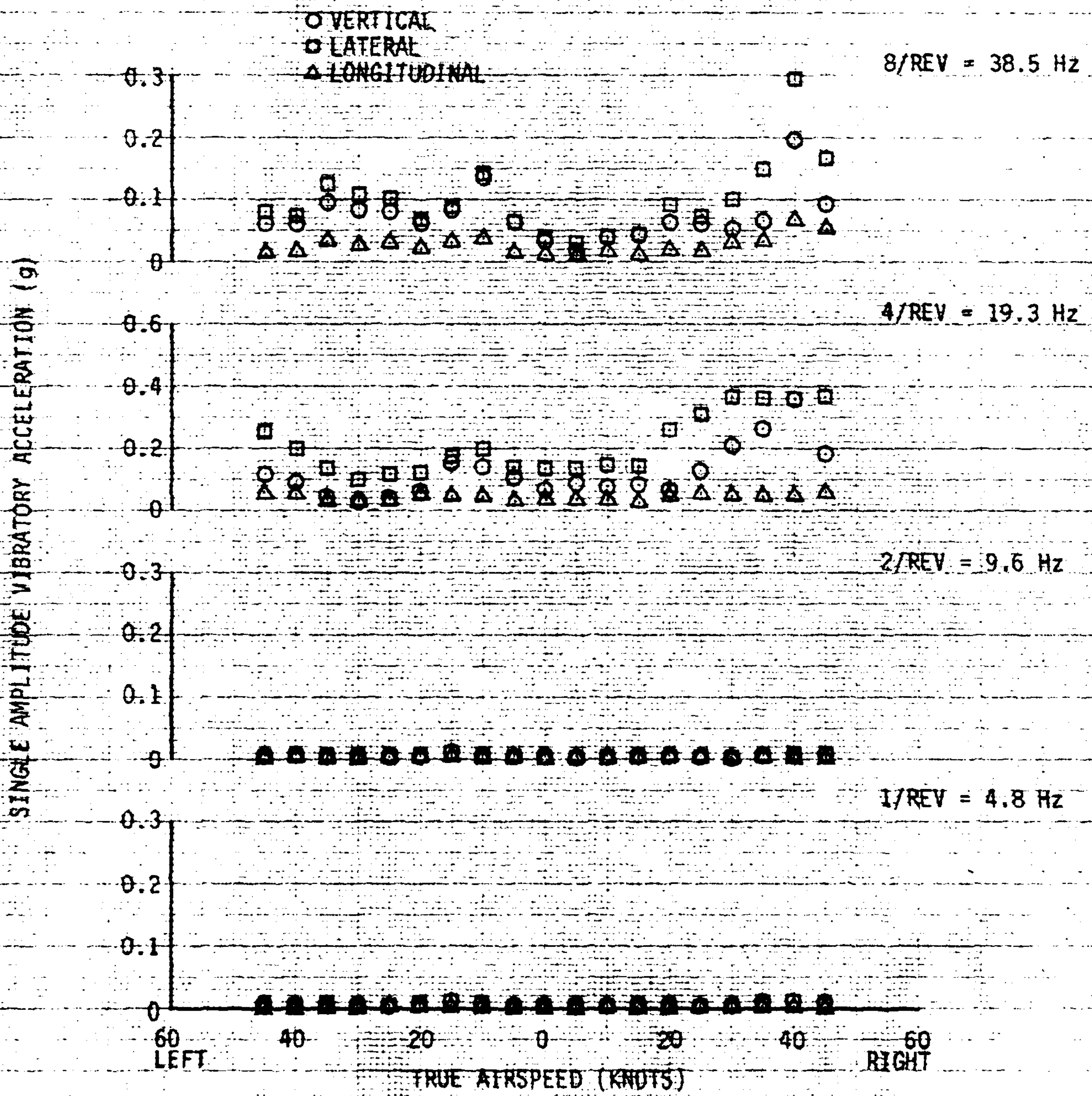


FIGURE 67
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14440	202.5 (FWD)	0.0 (MID)	10600	4.5	289	LOW SPEED

NOTES: 1. COPILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

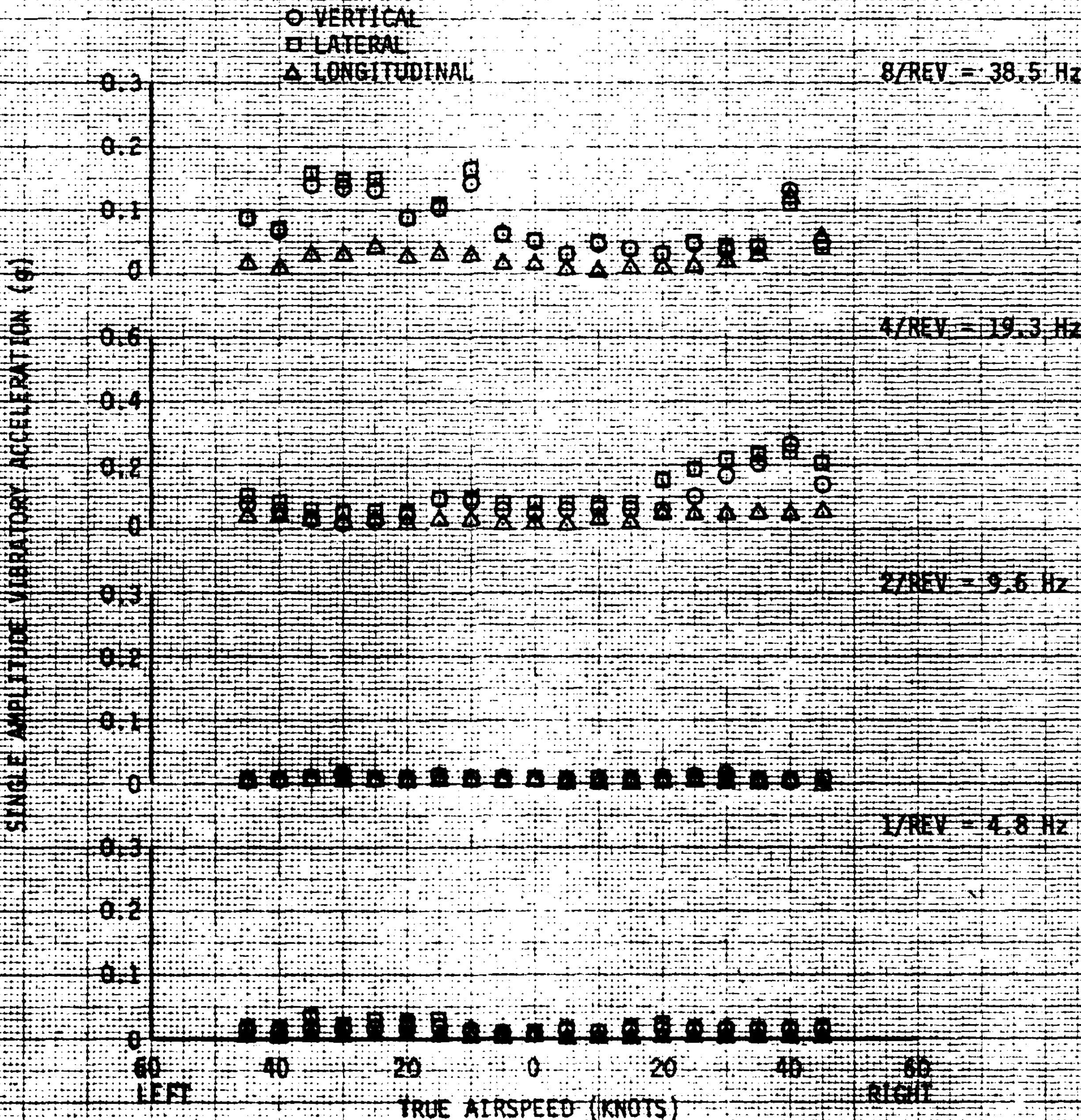


FIGURE 68
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
16080	201.8 (FWD)	0.0 (MID)	5800	21.5	290	LOW SPEED

NOTES: 1. PILOT SEAT
2. 16-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

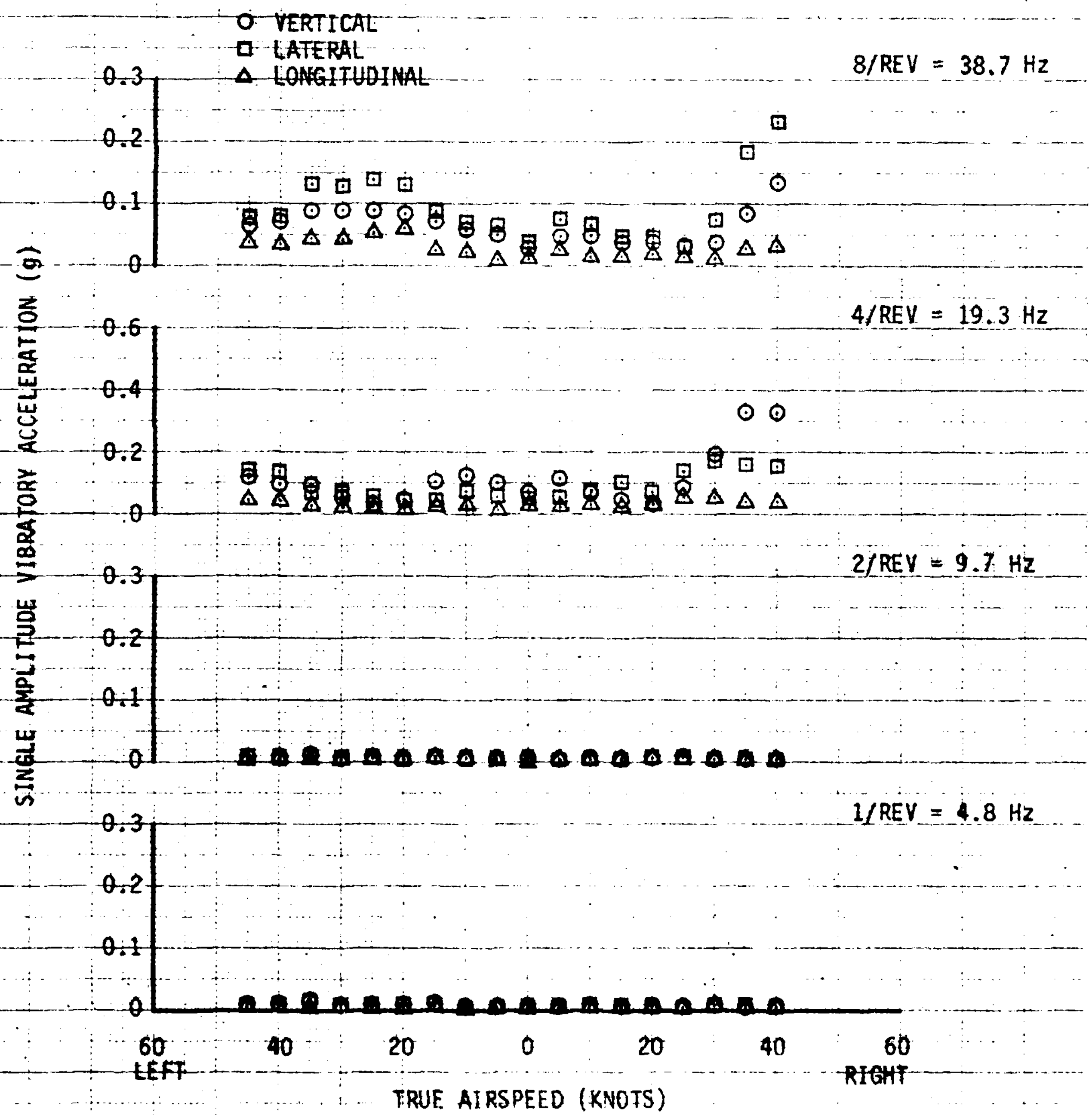


FIGURE 69
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
16080	210.8 (FWD)	0.0 (MID)	5800	21.5	290	LOW SPEED

NOTES: 1. COPILOT SEAT
2. 16-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET

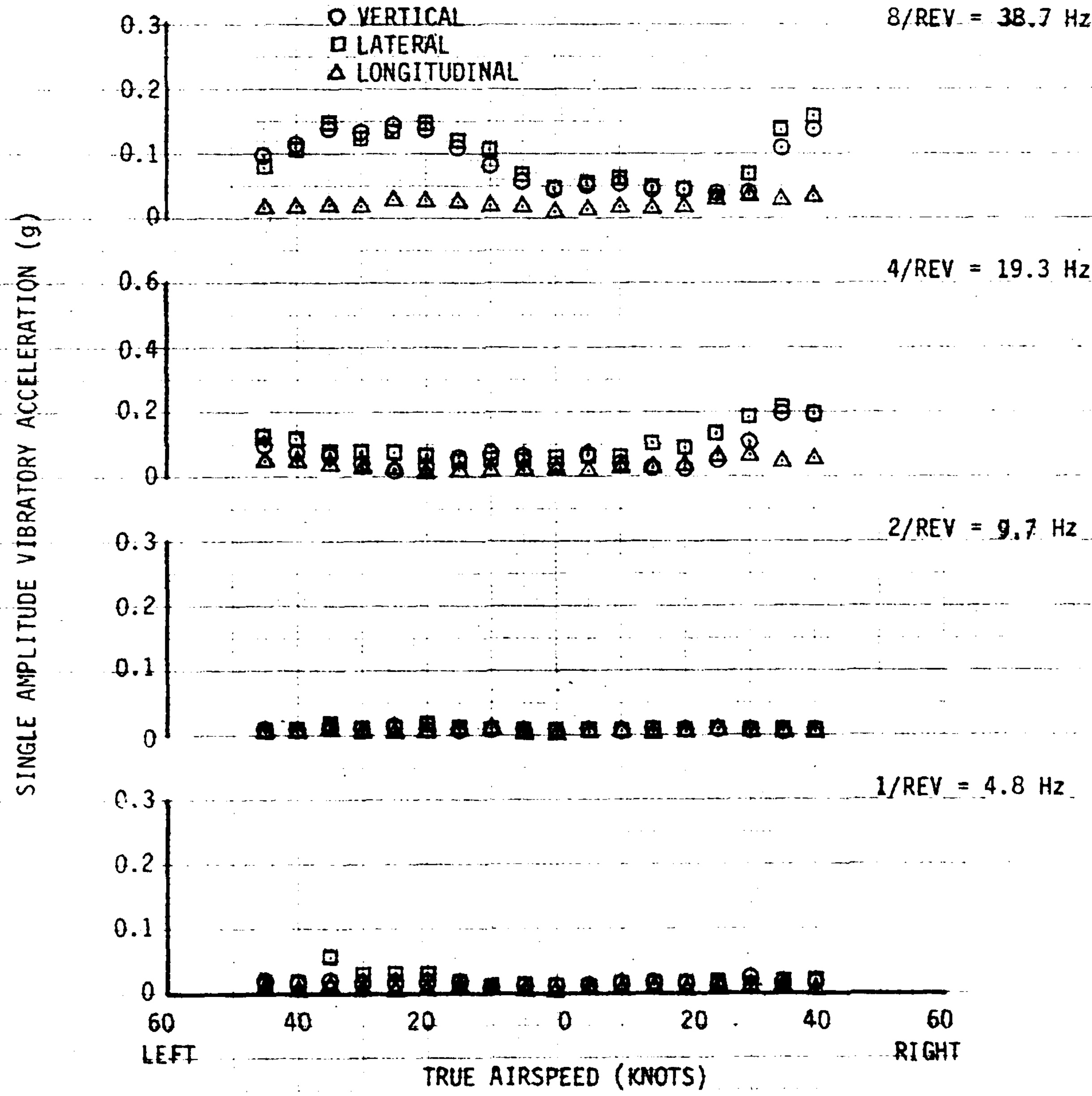


FIGURE 70
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14960	202.0 (FWD)	0.0 (MID)	5620	29.5	290	LEVEL

NOTES: 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION

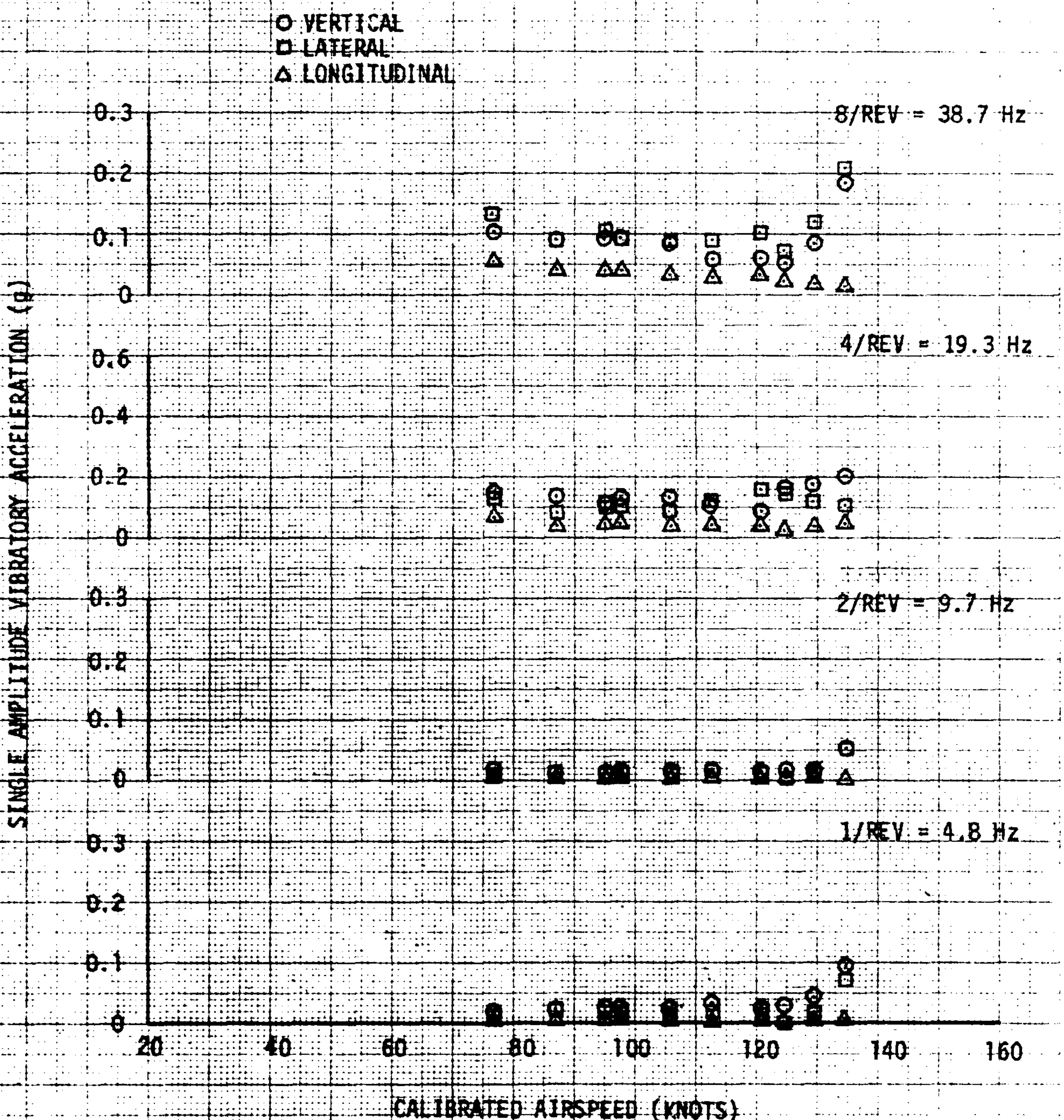


FIGURE 71
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
14960	202.0 (FWD)	0.0 (MID)	5620	29.5	290	LEVEL

NOTES: 1. COPILOT SEAT
2. 8-HELLFIRE CONFIGURATION

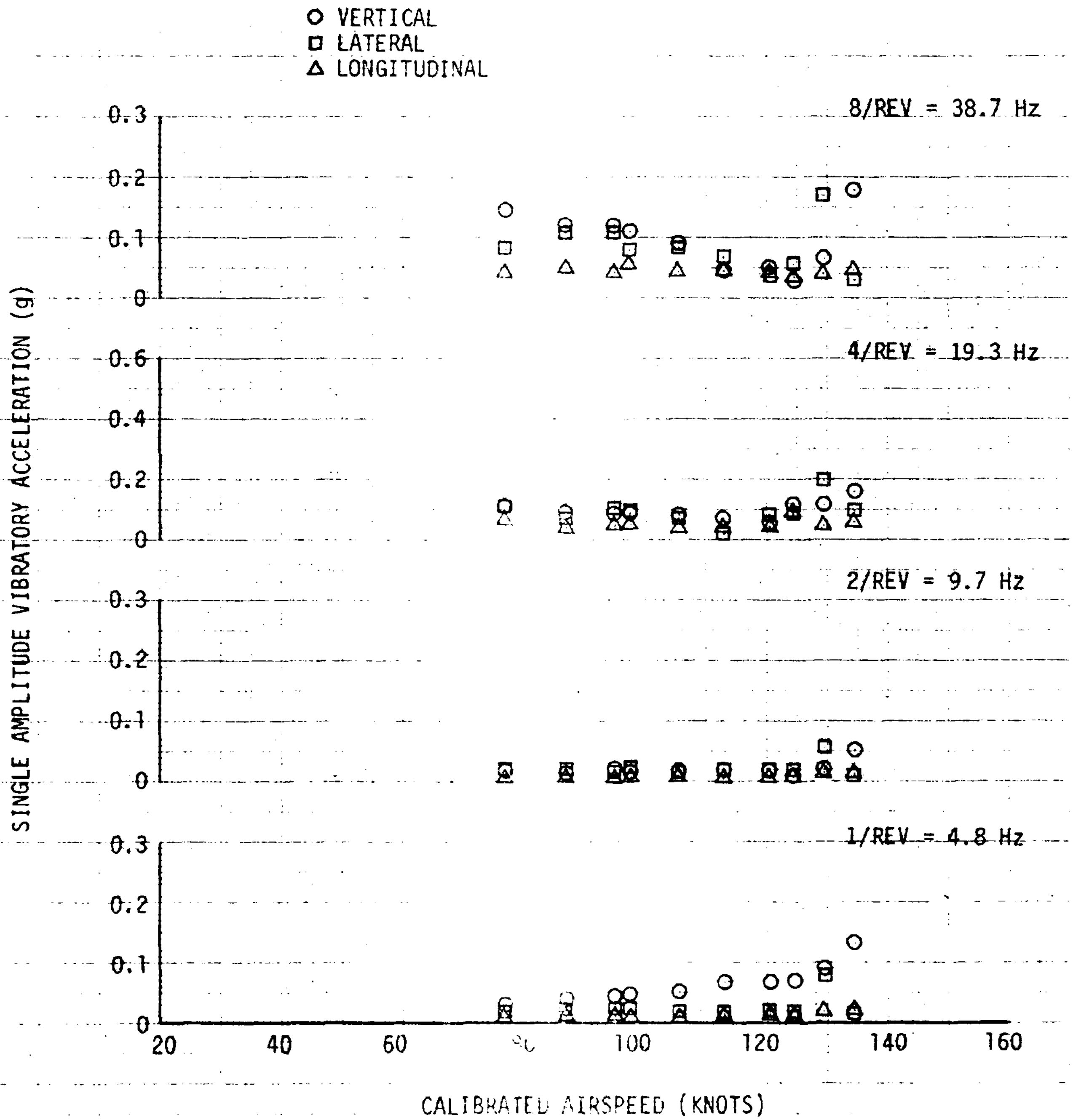


FIGURE 72
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15020	202.5 (FWD)	0.0 (MID)	5020	20.5	289	LEVEL

NOTES: 1. PILOT SEAT.
2. 8 HELLFIRE CONFIGURATION

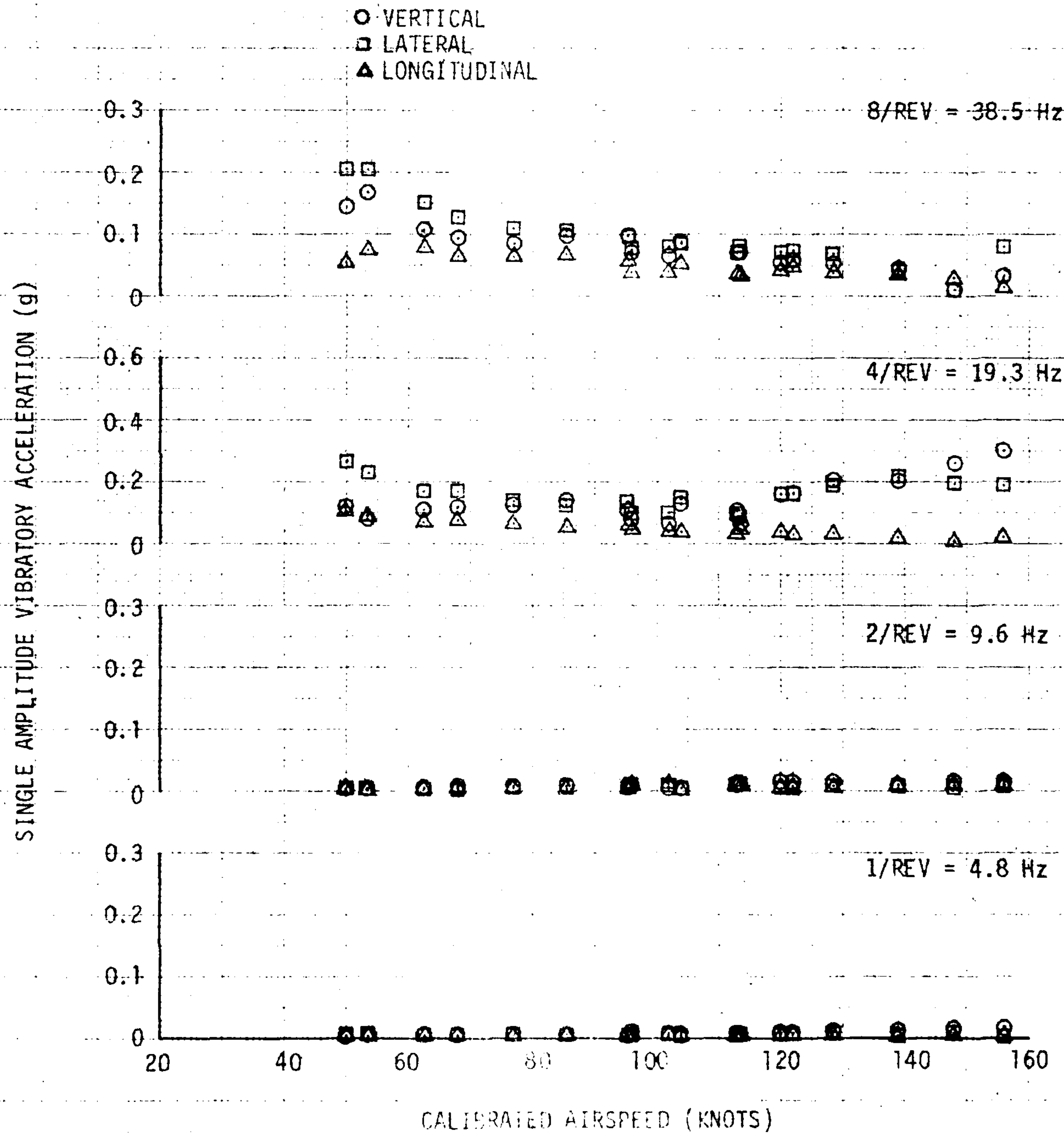


FIGURE 73
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15020	202.5 (FWD)	0.0 (MID)	5020	20.5	289	LEVEL

NOTES: 1. COPILOT SEAT
2. 8 HELLFIRE CONFIGURATION

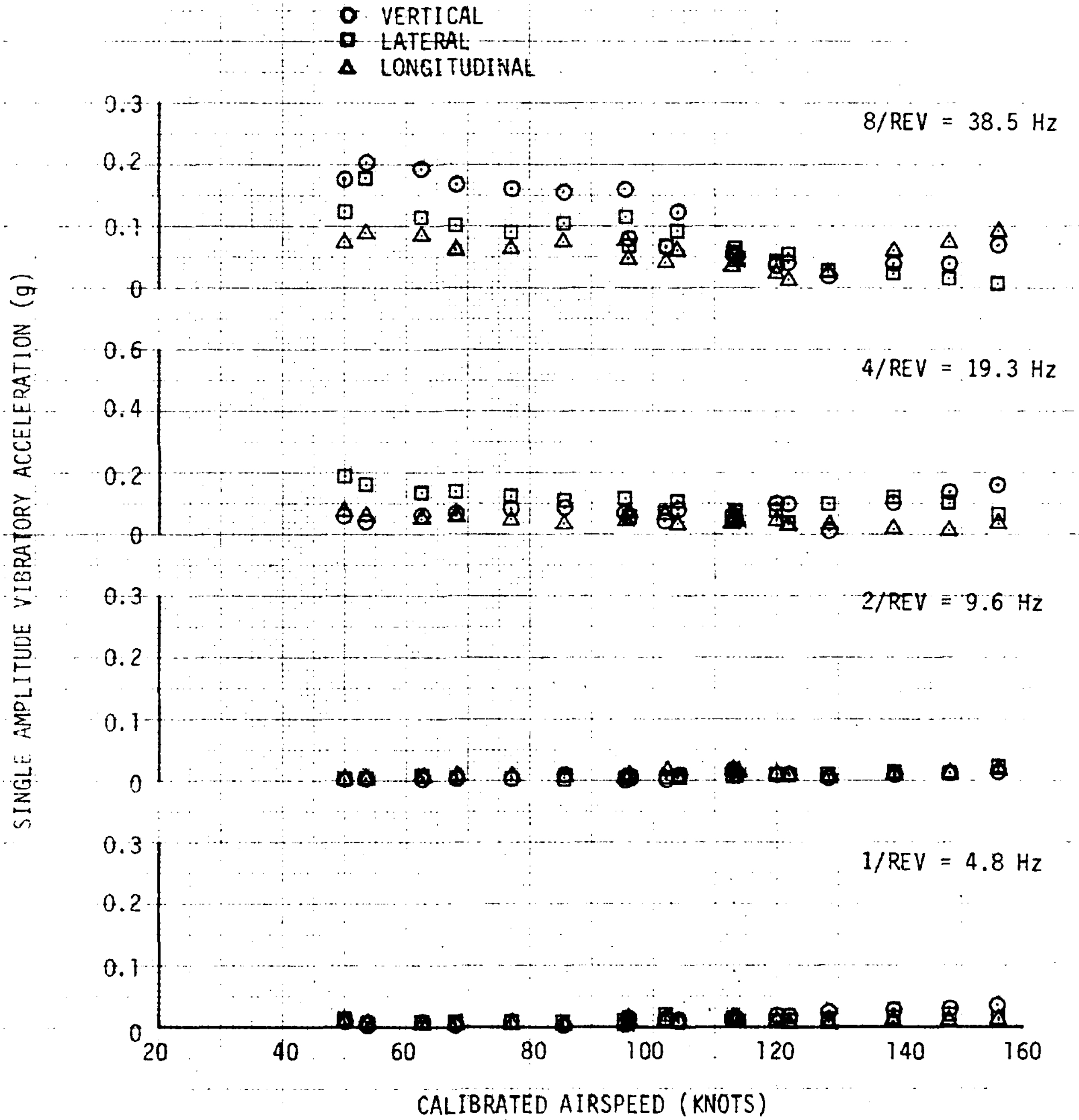


FIGURE 74
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15020	202.5(FWD)	0.0(MID)	5020	20.5	289	LEVEL

NOTES: 1. PILOT FLOOR
2. 8-HELLFIRE CONFIGURATION

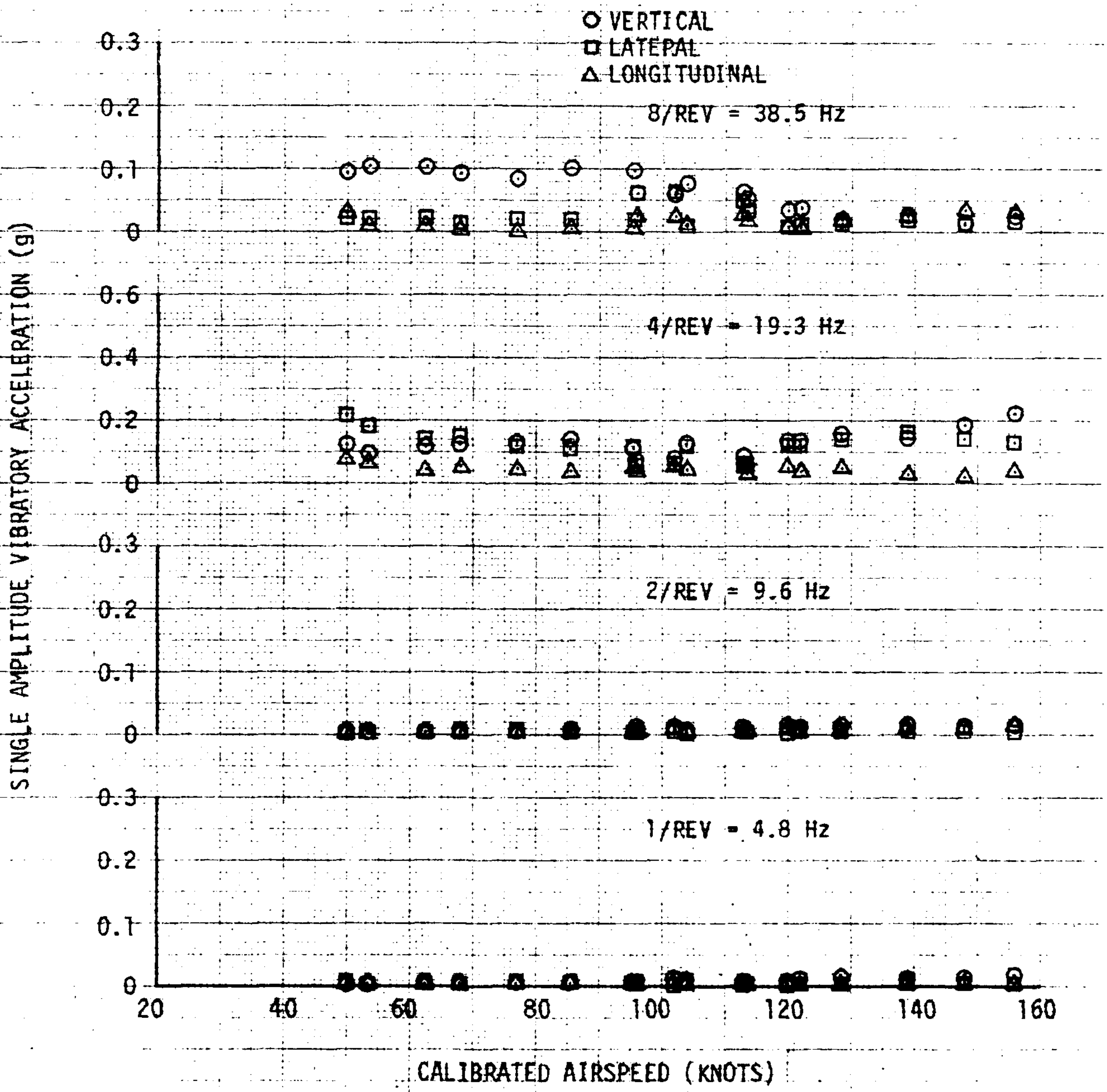


FIGURE 75
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15020	202.5 (FWD)	0.0 (MID)	5020	20.5	289	LEVEL

NOTES: 1. COPILOT FLOOR
2. 8 HELLFIRE CONFIGURATION

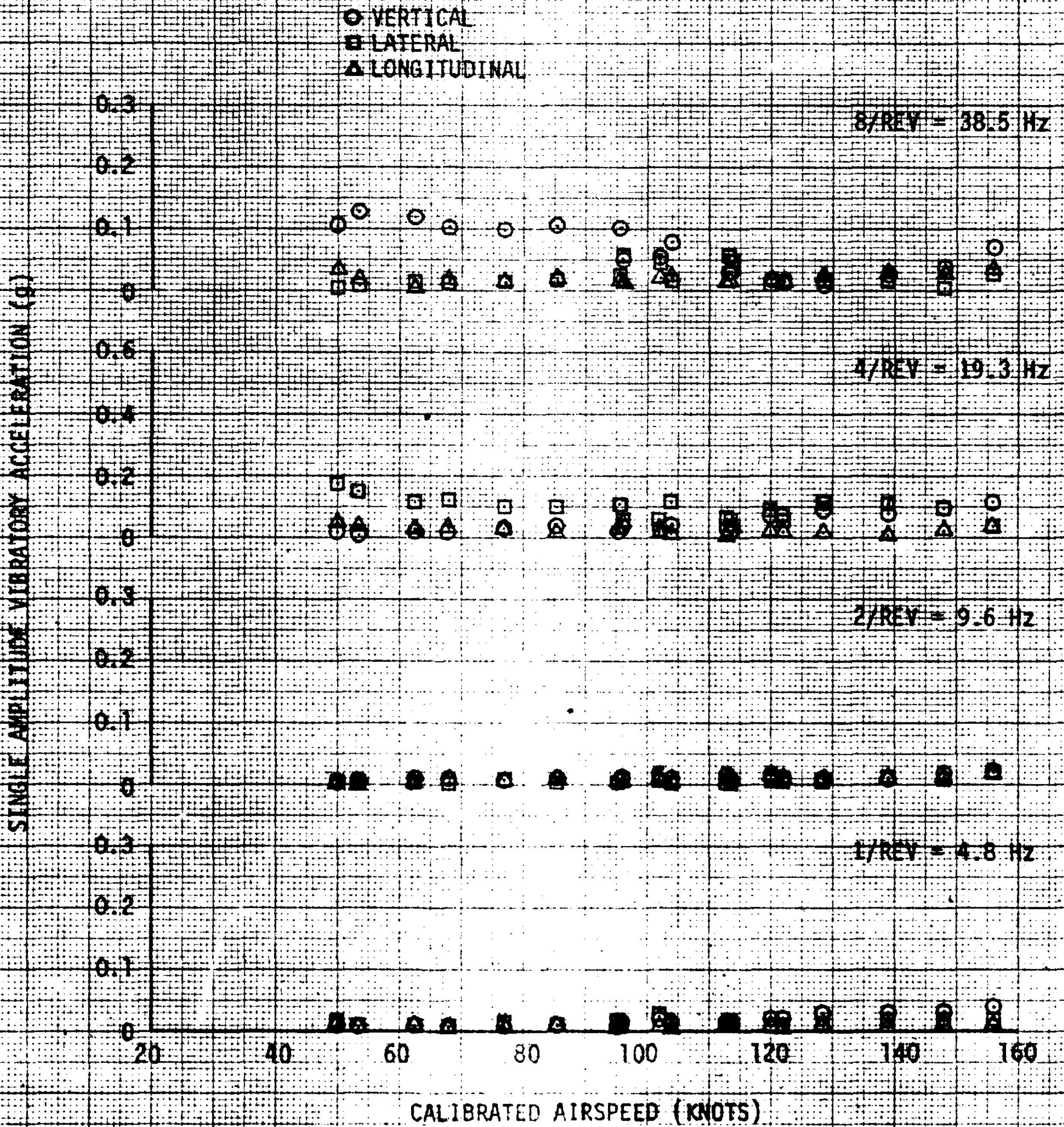


FIGURE 76
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23251

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
15020	202.5 (FWD)	0.0 (MID)	5020	20.5	289	LEVEL

NOTES: 1. AIRCRAFT CG
2. 8 HELLFIRE CONFIGURATION

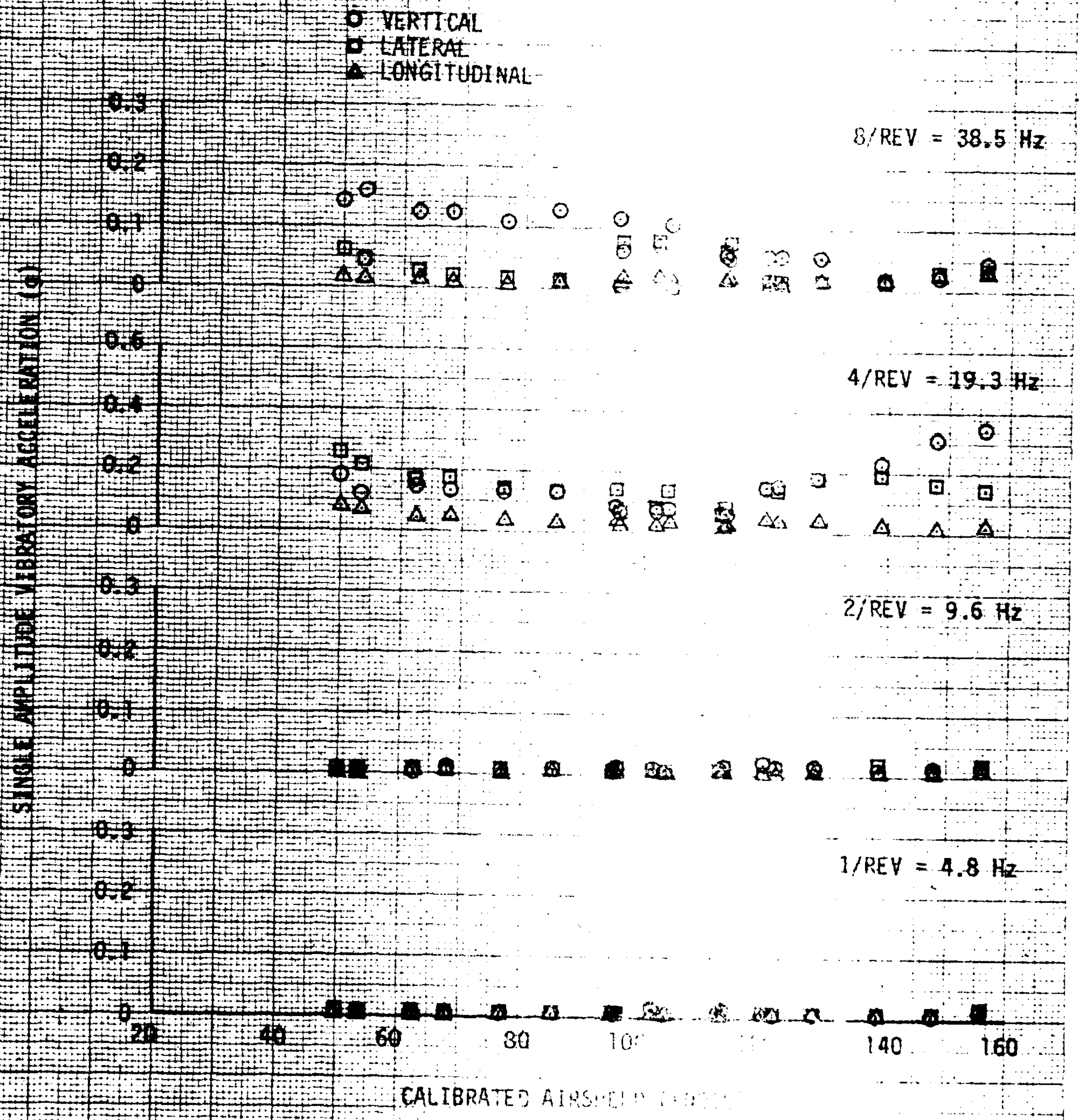


FIGURE 77
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15400	201.9 (FWD)	0.0 (MID)	13220	5.5	289	LEVEL

NOTES: 1. COPILOT SEAT
2. 8-HELLFIRE CONFIGURATION

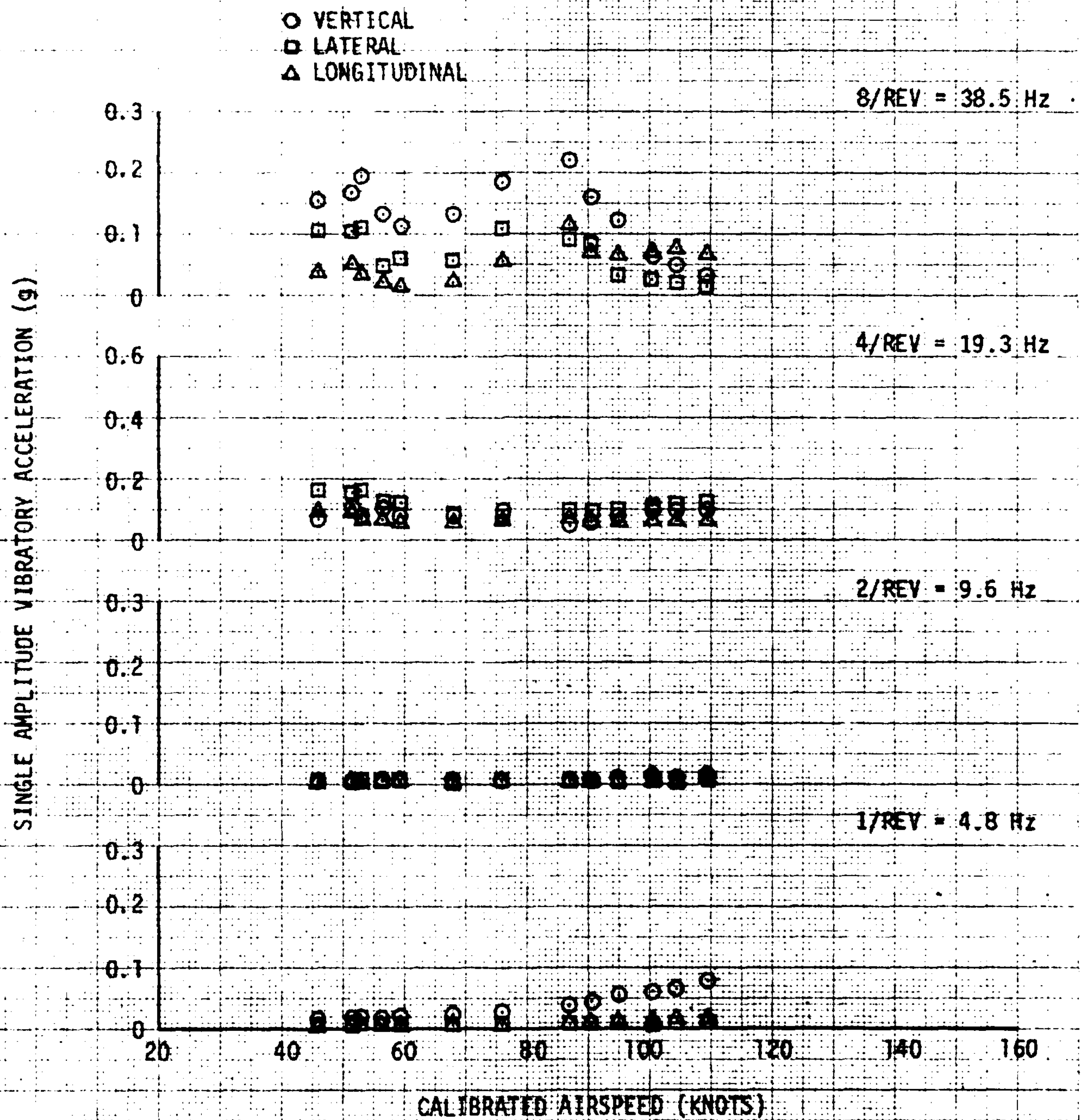
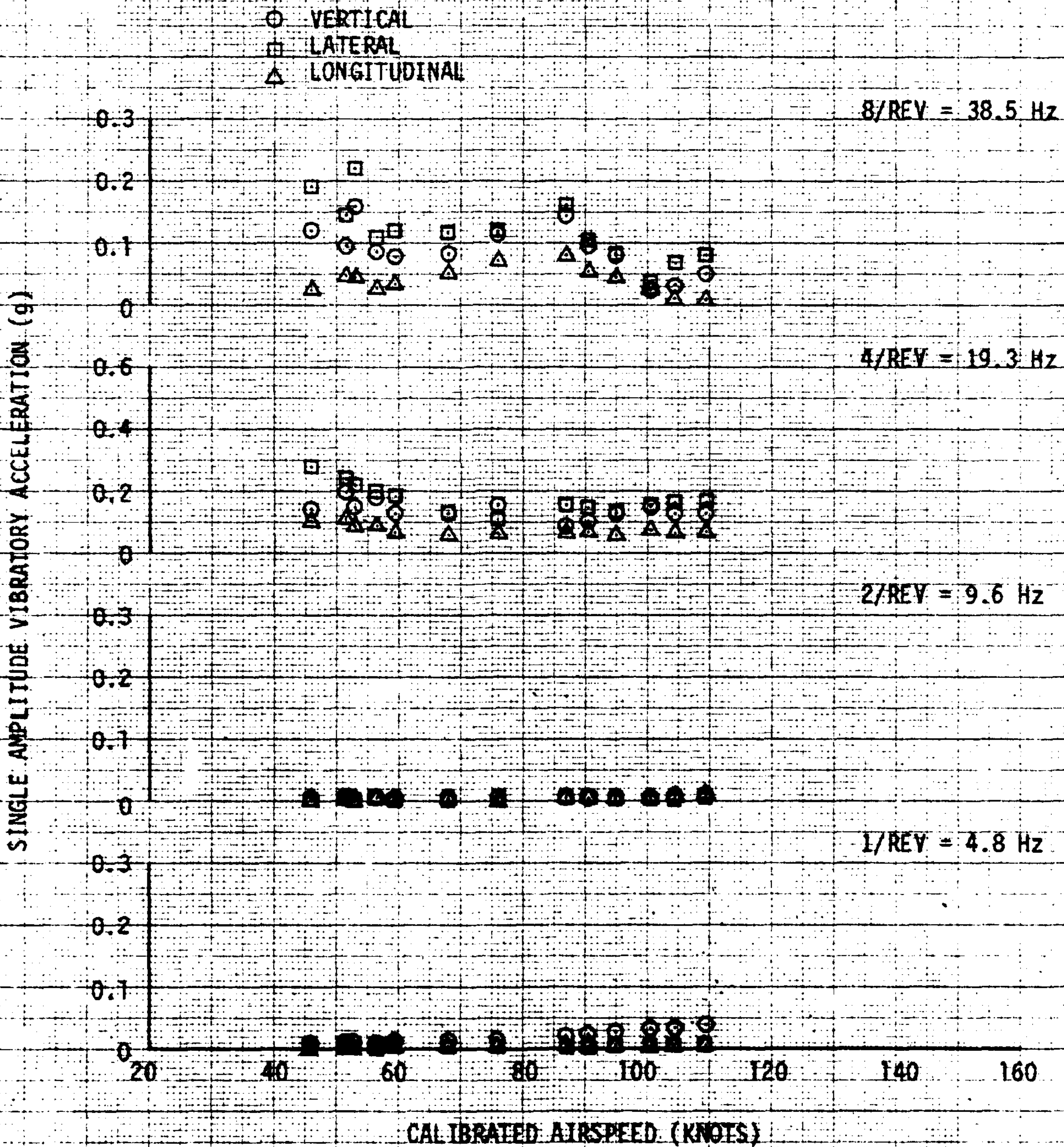


FIGURE 78
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15400	210.9 (FWD)	0.0 (MID)	13220	5.5	289	LEVEL

NOTES: 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION



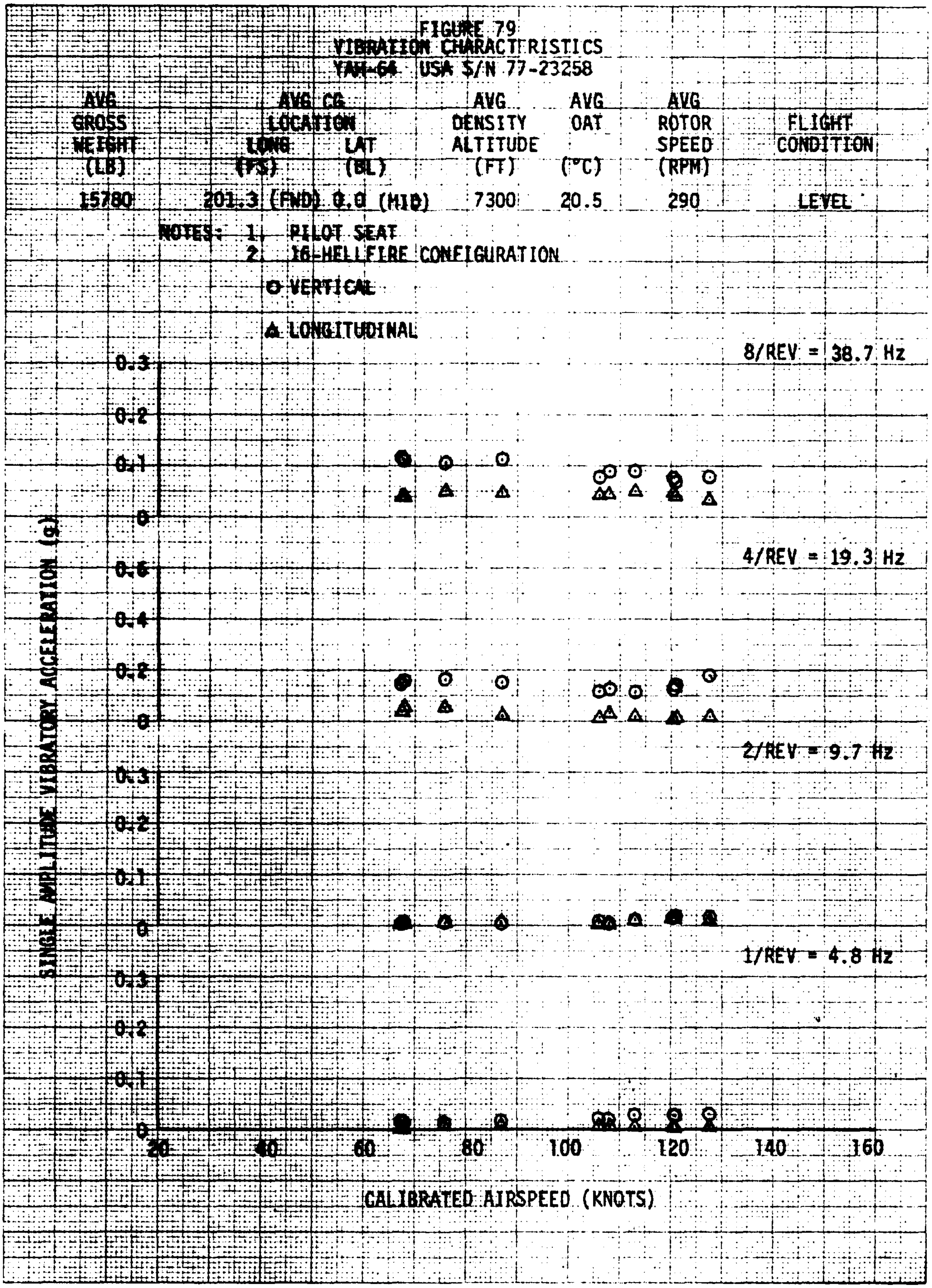


FIGURE 79
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15780
201.3 (FWD)
0.0 (MID)
7300
20.5
290
LEVEL
NOTES: 1. PILOT SEAT
2. 16-HELLFIRE CONFIGURATION
VERTICAL
LONGITUDINAL
8/REV = 38.7 Hz
4/REV = 19.3 Hz
2/REV = 9.7 Hz
1/REV = 4.8 Hz
SINGLE AMPLITUDE VIBRATORY ACCELERATION (g)
0.3
0.2
0.1
0
0.6
0.4
0.2
0
0.3
0.2
0.1
0
0.3
0.2
0.1
0
20
40
60
80
100
120
140
160
CALIBRATED AIRSPEED (KNOTS)

FIGURE 80
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

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
15780	201.3 (FWD)	0.0 (MID)	7300	20.5	290	LEVEL

NOTES: 1. COPILOT SEAT
2. 16-HELLFIRE CONFIGURATION

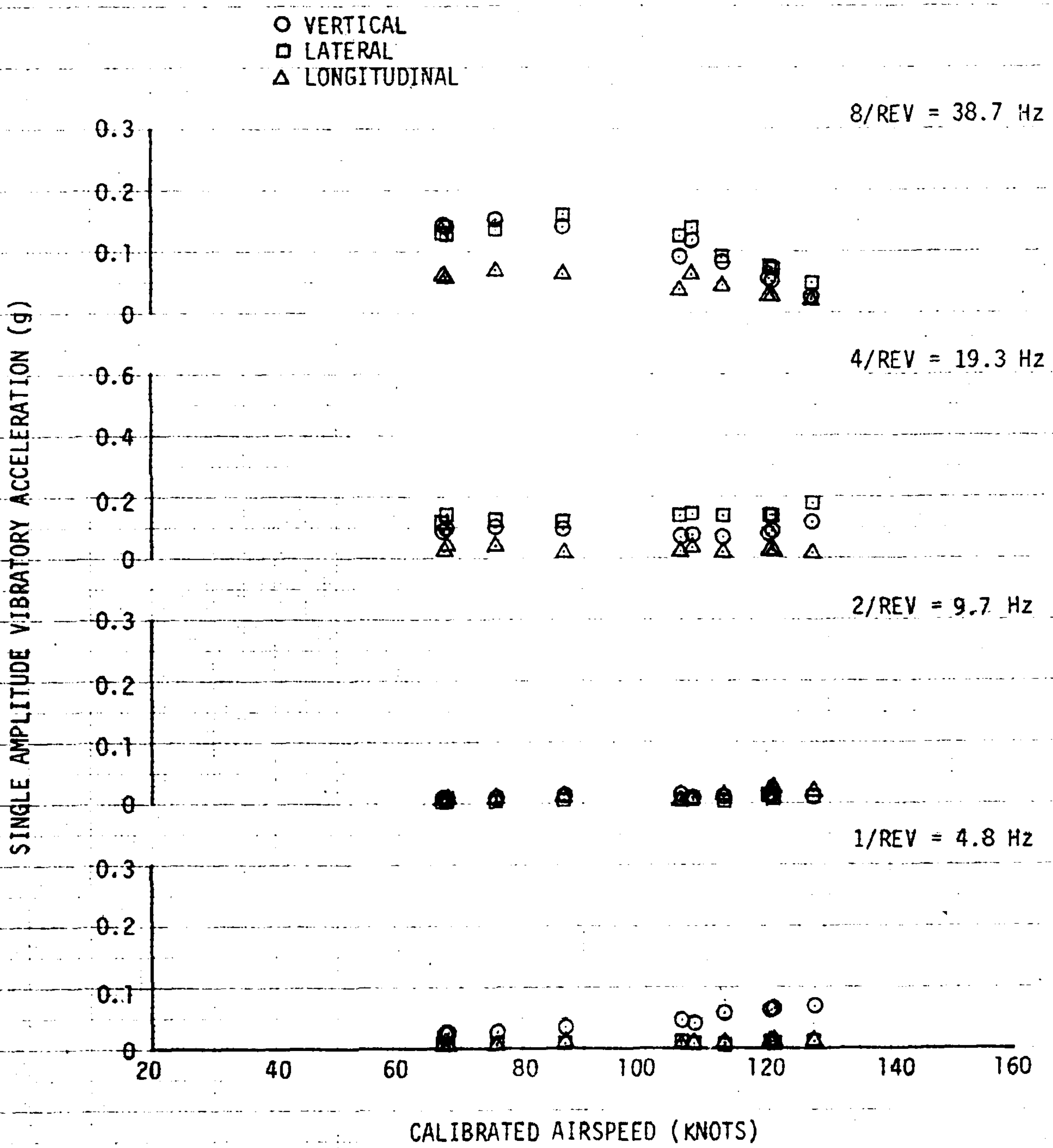


FIGURE 81
SHIP'S AIRSPEED CALIBRATION RIGHT WING
YAH-64 USA S/N 77-23258

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
15020	202.5 (FWD)	0.0 (MID)	5020	20.5	289	LEVEL FLIGHT

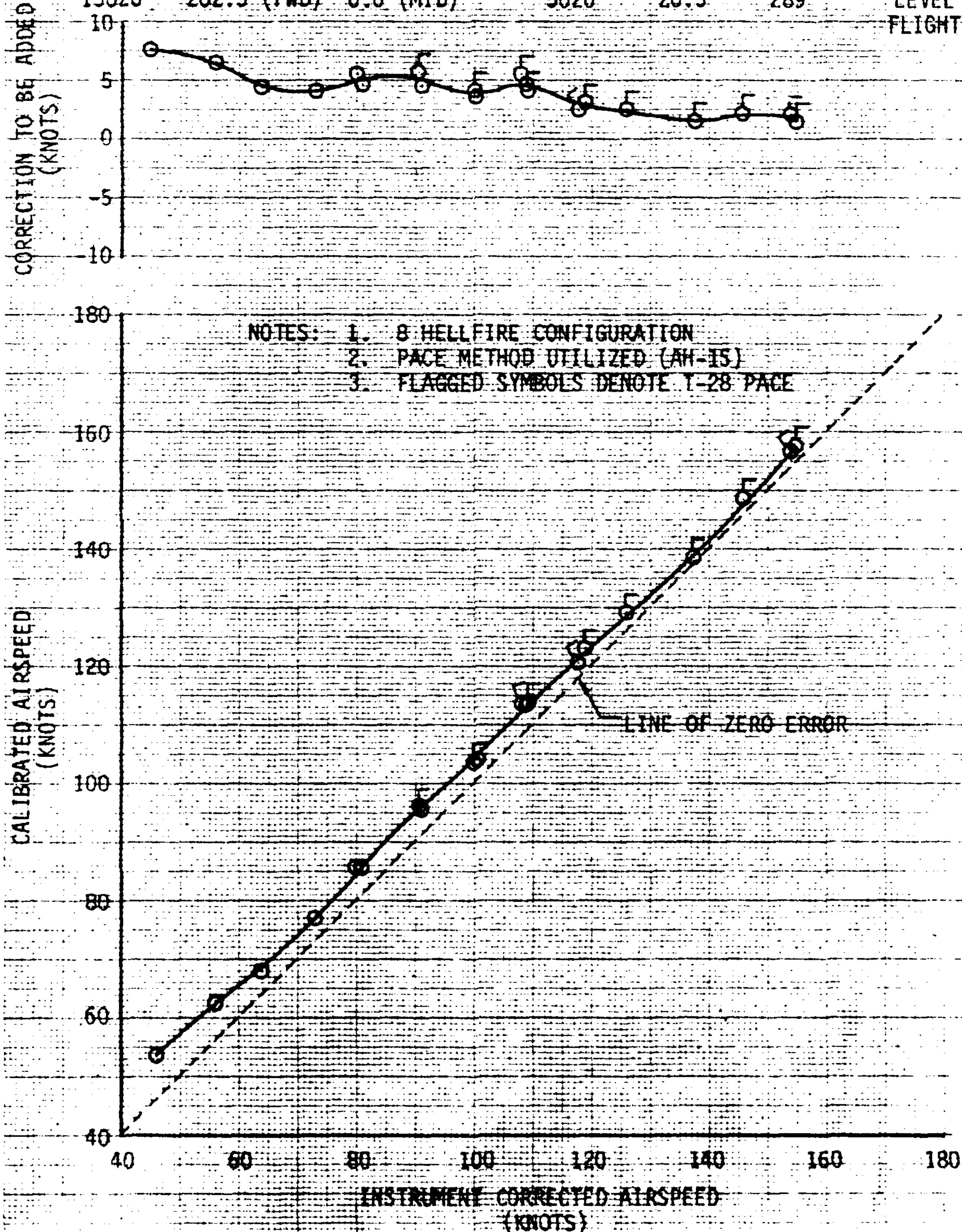


FIGURE 82
SHIP'S AIRSPEED CALIBRATION LEFT WING
YAH-64 USA S/N 77-23258

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
15020	202.5 (FWD)	0.0 (MID)	5020	20.5	289	LEVEL FLIGHT

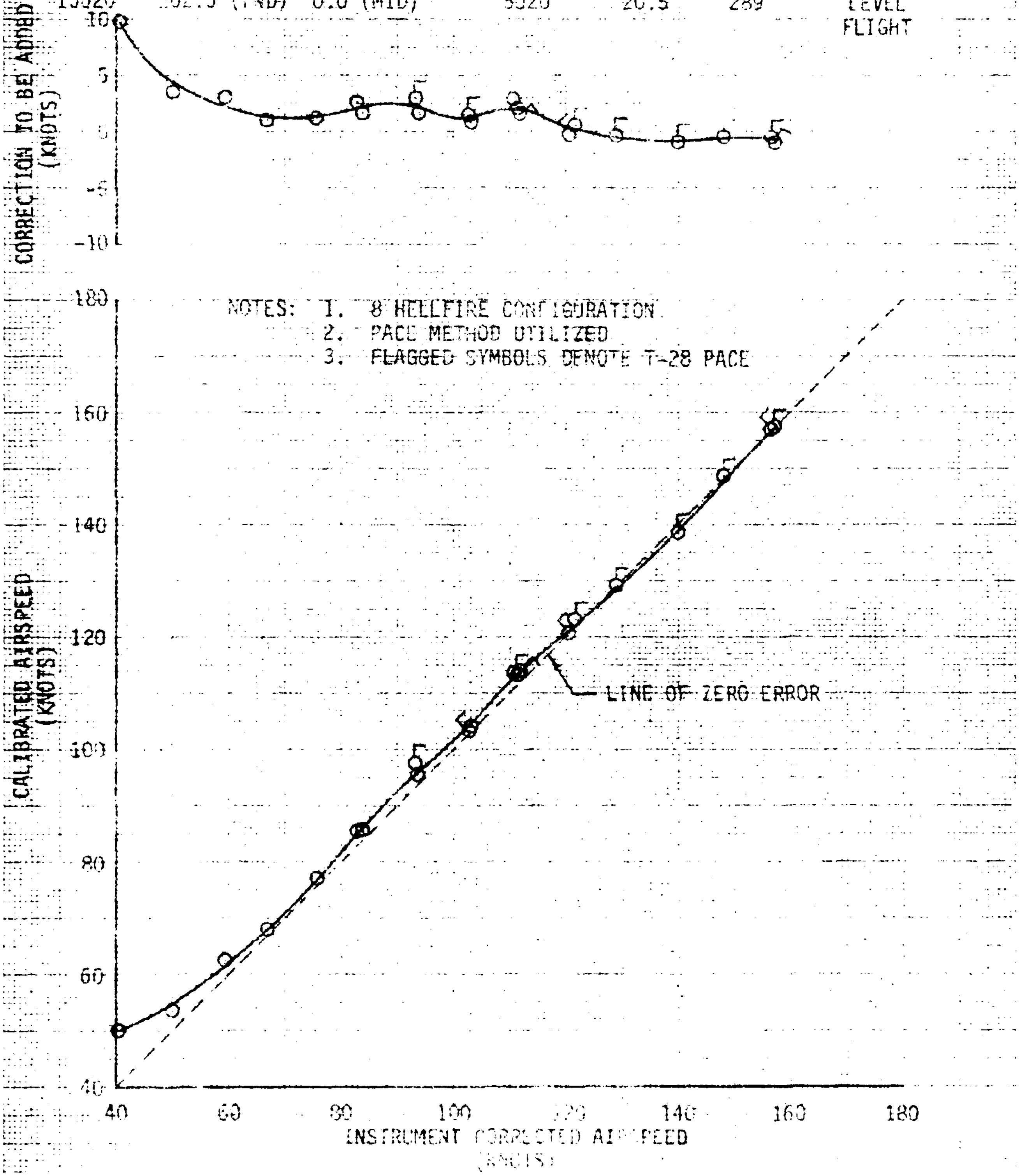


FIGURE 83
AIR DATA SYSTEM
LONGITUDINAL AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258

NOTES:
1. ADS = AIR DATA SYSTEM
2. TESTS CONDUCTED AT 20-FOOT WHEEL HEIGHT USING AN AUTOMOBILE FOR PACE
3. 8-HELLFIRE CONFIGURATION

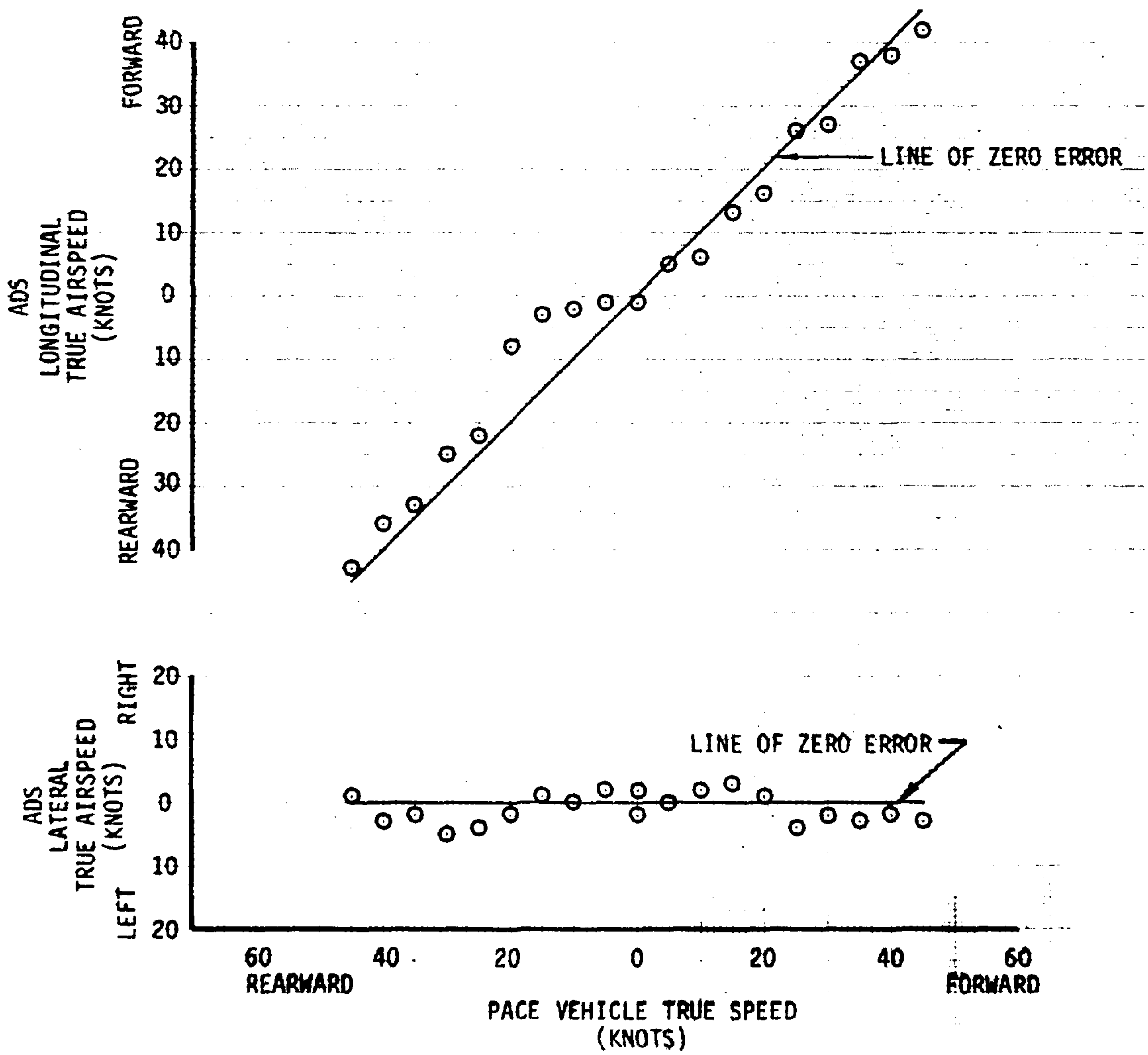
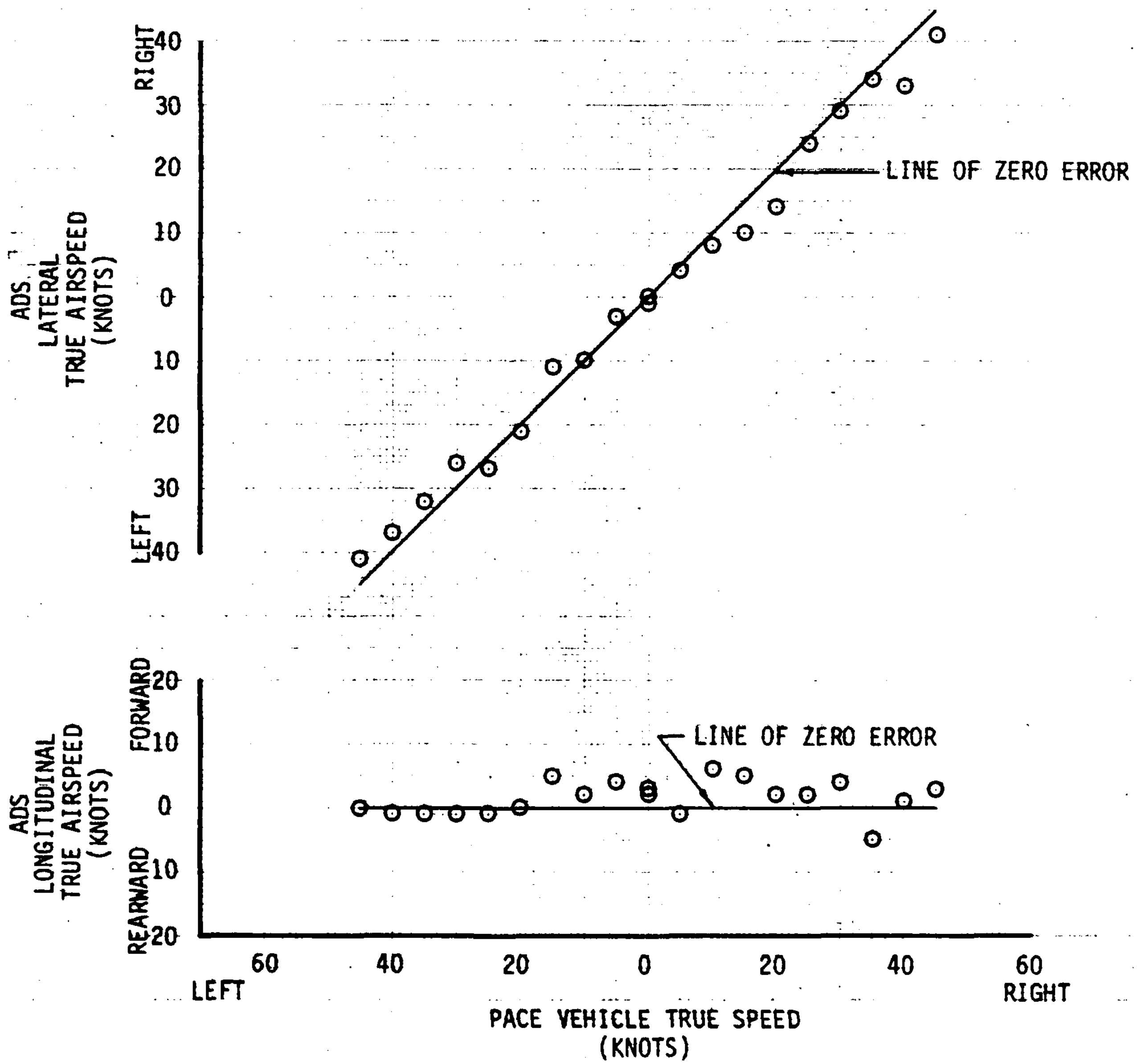


FIGURE 84
AIR DATA SYSTEM
LATERAL AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258

NOTES:
1. ADS = AIR DATA SYSTEM
2. TESTS CONDUCTED AT 20-FOOT WHEEL HEIGHT USING AN AUTOMOBILE FOR PACE
3. 8-HELLFIRE CONFIGURATION



APPENDIX F. ABBREVIATIONS

a	Speed of sound
A	Main rotor disc area (ft^2)
AAH	Advance Attack Helicopter
AC	Alternating Current
ADS	Air Data System
app	Appendix
APU	Auxiliary Power Unit
AVRADCOM	US Army Aviation Research and Development Command
A&FC	Airworthiness and Flight Characteristics
BL	Butt Line
BUCS	Back-up Control System
C	Celsius
CAS	Control Augmentation System
cg	Center of Gravity
C_L	Centerline
C_P	Coefficient of Power
CPG	Copilot/gunner
C_T	Coefficient of Thrust
DASE	Digital Automatic Stabilization Equipment
DC	Direct Current
deg	Degree
ECRR	Engineering Change Request and Record
ECU	Electrical Control Unit
EDT	Engineer Design Test
ENCU	Environmental Control Unit
EPR	Equipment Performance Report
ETP	Experimental Test Procedure
FABS	Forward Avionics Bays
fig.	Figure
fs, FS	Fuselage Station
ft	Feet
HAS	Hover Augmentation System
g	Acceleration of Gravity
GW	Gross Weight
HH	Hughes Helicopters
HMU	Hydromechanical Unit
HQRS	Handling Qualities Rating Scale
Hz	Hertz
IGE	In Ground Effect
IMC	Instrument Meteorological Conditions
in.	Inches
IR	Infrared
IRP	Intermediate Rated Power
KCAS	Knots Calibrated Airspeed
KIAS	Knots Indicated Airspeed
KTAS	Knots True Airspeed
LEU	Leading Edge Up
lb	Pound
LVDT	Linear Variable Displacement Transducer

M_{tip}	Advancing tip mach number
NAMPP	Nautical Air Miles Per Pound of Fuel
NOE	Nap of the Earth
N_P	Power turbine speed
N_R	Main rotor speed
OAT	Outside Air Temperature
OGE	Out of Ground Effect
PCM	Pulse Code Modulation
PIO	Pilot Induced Oscillation
PNVS	Pilot Night Vision System
psi	Pounds per Square Inch
Q	Engine output shaft torque
R	Radius (ft)
ref	Reference
RPM	Revolutions Per Minute
SAS	Stability Augmentation System
SCAS	Stability and Control Augmentation System
SCU	Stabilator Control Unit
sec	Seconds
SHP, shp	Shaft Horsepower
S/N	Serial Number
TADS	Target Acquisition and Designation System
TFS	Trim Feel System
TGT	Turbine Gas Temperature
USAAEFA	US Army Aviation Engineering Flight Activity
VDC	Volts Direct Current
V_H	Maximum Horizontal Velocity
VMC	Visual Meteorological Conditions
VNE	Never Exceed Airspeed
V_T	True airspeed
VRS	Vibration Rating Scale
WL	Water Line
W_f	Fuel flow rate

Greek and Miscellaneous Symbols

Δ	Incremental change
μ	Advance ratio
ρ	Air density (slugs/ft^3)
σ	Air density ratio
Ω	Main rotor angular velocity (radians/sec)
$\sim$	Approximately
4/rev	4th harmonic of the main rotor

APPENDIX G. EQUIPMENT PERFORMANCE REPORTS

The following EPR's were submitted:

Number	Subject
80-17-1	Yaw SAS Hardovers
80-17-2	Engine Compressor Stall
80-17-3	AUP Operation (sea level)
80-17-4	ENCU Performance (inadequate)
80-17-5	Stabilator (automatic mode failure)
80-17-6	Pressurized Air System (engine start)
80-17-7	Tailwheel Assembly
80-17-8	Engine TGT Fluctuations
80-17-9	Engine Power Levers (rotor locked start)
80-17-10	Engine Cowling Support
80-17-11	Marconi Scale (N_P + N_R indications)
80-17-12	Engine Nose Gearbox Seals
80-17-13	IR Suppressor Assembly
80-17-14	Intermittent Master Cuation Light
80-17-15	Flight Control Rod Ends
80-17-16	SDC Manifold
80-17-17	Collective Friction
80-17-18	Stabilator Bearings (bushings)
80-17-19	Thermistor Installation (tail rotor gearbox)
80-17-20	Engine noise
80-17-21	Nitrogen Bottle Color
80-17-22	Tube and Line ID Tape

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Deputy Chief of Staff for 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-EA)	1
US Army Materiel Development and Readiness Command (DRCDE-SA, DRCQA-E, DRCRE-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)	3
US Army Combined Arms Center (ATZLCA-DM)	1
US Army Safety Center (IGAR-TA, IGAR-Library)	2
US Army Research and Technology Laboratories/Applied Technology Laboratory (DAVDL-AS/DAVDL-POM)	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-ATL-D)	1
US Army Research and Technology Laboratories/Propulsion Laboratory (DAVDL-PL-D)	1
US Army Materiel Development and Research Command (DALO-AV)	1
US Military Academy (MADN-F)	1
Defense Technical Information Center (DDR)	12
Program Manager (DRCPM-AAH-SE, DRCPM-AAH-APM-TE)	6
US Army Operational Test and Evaluation Agency (CSTE-TM-AV)	1
General Electric - AEG (Mr. Koon)	1
US Army Materiel System Analysis Agency (DRXSY-AAS)	1
Hughes Helicopter (Mr. Gerry Ryan)	3
US Army Missile Command (DRCPM-HDT-T)	1