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FIRST ARTICLE PREPRODUCTION TESTS OF THE AH - 64A HELICOPTER

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FINAL REPORT

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

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First article preproduction flight testing was conducted in Mesa, Arizona (elevation 1387 feet) between 6 August and 29 August 1984. Twenty test flights were conducted for a total of 32.2 hours (22.1 hours productive). Level flight performance and handling qualities tests were conducted to verify that the production aircraft characteristics are the same as those of prototype YAH-64 as tested in Airworthiness and Flight Characteristics (Part 3) tests. The level flight performance and handling qualities of the production

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aircraft were not significantly different from those of the prototype. Additionally, the performance data presented in the operator's manual is a good representation of the production aircraft performance. One enhancing characteristic was found (the automatic contingency power feature of the T700-GE-701 engine). Two deficiencies were identified (the false indication of engine failures which was previously reported and the poor engine/airframe response characteristics which was previously reported as a shortcoming). Four shortcomings of the prototype aircraft have been corrected in the production aircraft while 13 remain. Additionally, 13 new shortcomings were found.

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SUBJECT: ERRATA Sheet, USAAEFA Project No. 84-10, First Article Preproduction Tests of the AH-64A Helicopter.

1. Paragraph 2c of the AVSCOM Position Letter (located just after Table of Contents) was left out of the printed report. That paragraph should read as follows:

c. Paragraph 60a - The 4/rev vertical vibration has been reported in several previous reports, and the Directorate maintains its concurrence in the shortcoming.

2. This ERRATA Sheet should be included in all copies of the subject report.

DEPARTMENT OF THE ARMY
HEADQUARTERS, US ARMY AVIATION SYSTEMS COMMAND
4300 GOODFELLOW BOULEVARD, ST. LOUIS, MO. 63120-1798

REPLY TO
ATTENTION OF

AMSAV-E

SUBJECT: Directorate for Engineering Position on the Final Report of USAAEFA Project No. 84-10, First Article Preproduction Tests of the AH-64A Helicopter

SEE DISTRIBUTION

1. The purpose of this letter is to establish the Directorate for Engineering position on the subject report. The objectives of this test were to obtain helicopter handling qualities and performance data on the first production AH-64A and compare these data to those obtained during previous tests of the prototype AH-64 helicopter.

2. This Directorate agrees with the report conclusions and recommendations, with the exceptions identified herein. Conclusions and recommendations are discussed by paragraph as indicated.

a. Paragraph 59a - The contractor has developed a correction to the false indications of an engine failure. Engine torque is now an input signal to the engine out warning box along with engine rpm. This new software has been bench tested and flight tested satisfactorily. These actions correct this deficiency.

b. Paragraph 59b - The undesirable engine/airframe response was previously reported under AEFA Project No. 80-17-3 as a shortcoming; however, at that time the full effects on NOE flight were not realized. This Directorate now agrees that the reported characteristic was a deficiency. In an effort to correct this deficiency, the engine and airframe contractors made modifications to the load demand spindle (LDS) bellcrank, engine electronic control unit (ECU), hydromechanical unit (HMU), and added a collective anticipator system. These modifications were flight tested by AEFA pilots subsequent to this report and have satisfactorily corrected this deficiency. A formal report of the correction and flight evaluation is forthcoming.

d. Paragraph 60b - This Directorate agrees that the restricted pilot's field-of-view meets the definition of a shortcoming; however, this item is more truely the subject of the operational evaluation. It is important that qualitative assessments of the helicopter be made, but the degree of acceptability should remain with the operational evaluators.

e. Paragraph 60c - Access to the pilot's #1 fire handle will be improved by eliminating the offending floodlight with the incorporation of blue-green lighting.

f. Paragraph 60d - This directorate agrees that inconsistent HARS alignment accuracy is a shortcoming. HARS alignment accuracy has been demonstrated to be within specification requirements, however, the AAH mission requires dependable navigation, ie - consistent alignment accuracy. The faults cited here could have been due to hardware or software errors. Improvements in these areas are currently being investigated by McDonnell Douglas Helicopters.

g. Paragraph 60e - The engine out warning (EOW) box circuitry used during these tests has been modified to prevent the high rotor speed activation. The new modified EOW box has been evaluated during subsequent flights and satisfactorily corrects this shortcoming.

h. Paragraph 60f - The Government has directed a split light on the fuel control panel to show both (1) a positive indication of fuel transfer and (2) a positive indication of crossfeed in progress. The contractor is preparing an ECP for incorporation of this fix. Upon incorporation, this ECP is expected to correct this shortcoming.

i. Paragraph 60g - AVSCOM agrees that the design of the pilot's cyclic grip is a shortcoming. The contractor is investigating relocating the HAS switch to the cyclic grip. The Government anticipates an ECP for relocation of the HAS switch upon successful completion of this investigation.

j. Paragraph 60h - The oil leakage that results in this accumulation is being remedied in production by changes to the sealant procedures and using a different "O" ring.

k. Paragraph 60i - The grease expulsion from the intermediate gearbox is being corrected by adding a hole in the baffle. An 80 hour test program has demonstrated the effectiveness of this fix. An ECP has been approved by the Government for incorporation in all aircraft, including retrofit. Additionally, a letter has been issued notifying users that these units must be shipped and handled in an upright position.

l. Paragraph 60j - This Directorate agrees that the inconsistent switch labeling between pilot and CPG fuel panels is a shortcoming and the Government has initiated action on this item. We also agree that the design of the pilot fuel control panel is a shortcoming. A redesign of the fuel control panel was agreed to by the Army and the contractor that called for the crossfeed and fuel ON/OFF switches to be of different design. While these new switches are different in appearance, they are indistinguishable by feel, especially while wearing gloves. The redesign efforts in this area are continuing.

m. Paragraph 60k - This Directorate does not agree that this is a shortcoming. When the longitudinal trim system is activated during commanded attitude changes with attitude hold ON the SAS system recenters the SAS longitudinal actuator. This is required for the attitude hold to operate and it operates in accordance with the system design. No action is planned to change the system operation.

n. Paragraph 601 - AVSCOM agrees with this finding and the main rotor system plunger assembly has been deleted from the production helicopter. A visual inspection procedure has been developed and is being used instead.

o. Paragraph 60m - Slack in the parking brake actuating cable allows the handle to be pushed in without releasing the parking brake. The Government is reviewing alternate designs and will request an ECP to correct this shortcoming.

p. Paragraph 60n - AVSCOM does not agree that this is a shortcoming. The crashworthy design features preclude additional longitudinal adjustment or seat padding of the pilot's seat. A lumbar support has been added to increase support but no additional action is planned.

q. Paragraph 60o - The total concept of the anti-ice and caution-warning panels is being reviewed by the Government and the contractor, with the intent of redesigning both. These efforts are intended to resolve this shortcoming.

r. Paragraph 60p - The location of the pilot engine control quadrant is not considered a shortcoming. The layout and location of this quadrant was carefully evaluated during design reviews and mock-up reviews. The quadrant is located as close as allowed with required collective control clearance. No action is planned.

s. Paragraph 60q - AVSCOM agrees that this is a shortcoming and the Program Management Office (PMO) has directed correction by the contractor. A redesign is currently being bench tested by the contractor. Army pilot evaluation will follow the contractor's tests.

t. Paragraph 60r - The location of the tailwheel lock/unlock light is not considered a shortcoming. The location of this light was evaluated during Government design and mock-up reviews and is acceptable in its present location.

u. Paragraph 60s - AVSCOM agrees that this is a shortcoming. The engine power lever assembly in the test helicopter had been incorrectly assembled and the cables were damaged during installation. The engine power lever control cable assembly has been redesigned to avoid this situation in the future. The redesign has been evaluated and approved by Army pilots and will be incorporated in the production helicopter (ECP 694).

v. Paragraph 60t - The background noise has been eliminated through redesign and improvement of the grounding in the strobe power supply. The production test procedures have also been modified to insure optimum ADF alignment. No further action is planned.

w. Paragraph 60u - AVSCOM agrees that the lack of VDU roll symbology to show up/down direction is a shortcoming. The Government is initiating action to change the VDU display.

x. Paragraph 60v - An ECP (ECP 421R1) has been initiated to provide independent control of the CPG Marconi indicators.

y. Paragraph 60w - AVSCOM agrees that poor readability of the AN/APX-100(V) transponder code is a shortcoming; however, the design and location have been optimized for the crew station and no further action will be taken.

z. Paragraph 60x - The production helicopters number 52 and subsequent engine oil access doors will be fitted with a wing type camlock fastener to replace the slotted screw, thereby correcting this shortcoming. This change will not be retrofitted to other helicopters.

aa. Paragraph 60y - The caution-warning panel is in review by both AVSCOM and the contractor for redesign. This effort includes software changes to the rocket control panel. These efforts are expected to resolve this shortcoming.

bb. Paragraph 60z - The EMI in the caution-warning panel has been corrected through redesign of the Marconi test circuit. However, the indications that accompanied selection of number one and number two generator test switches were the result of incorrect procedure. The operator's manual (-10) is being corrected to avoid future occurrences. The pylon articulation has been corrected by ECP 707, which modified the Textron ESCS logic.

FOR THE COMMANDER:

James A. Ray

DANIEL M. McENEANY
Director of Engineering

INTRODUCTION

BACKGROUND

1. The US Army contracted with Hughes Helicopters, Inc. (HHI) for production of the AH-64A Apache, Advanced Attack Helicopter. First article preproduction testing was required by the Production Airworthiness and Qualification Program contained in the system specification (ref 1, app A). The US Army Aviation Engineering Flight Activity (USAAEFA) has conducted several tests of the prototype YAH-64 (refs 2 through 9). In March 1984, the US Army Aviation Systems Command (AVSCOM) requested (ref 10) USAAEFA to conduct an evaluation of the first production aircraft to verify that the performance and handling qualities were not significantly different from those of the prototype.

TEST OBJECTIVES

2. The objective of this First Article Preproduction Test was to obtain quantitative flying qualities and performance data on the first article AH-64A. Additionally, a comparison was made of data obtained during this test to that gathered during the Airworthiness and Flight Characteristics Test, Part 3 (A&FC 3) (ref 9).

DESCRIPTION

3. The AH-64A (USA S/N 82-23355) is a two-place, tandem-seat, twin engine helicopter with four-bladed main and antitorque rotors and conventional wheel landing gear. The helicopter is powered by two General Electric T700-GE-701 turboshaft engines. The AH-64A has a movable horizontal stabilator with three modes of operation: Manual, Automatic, and nap-of-the-earth (NOE)/Approach. A dummy 30mm gun, the Target Acquisition and Designation System (TADS), and the Pilot Night Vision System (PNVS) were installed on the aircraft. The helicopter has a wing with two store pylons (shorter than those on the YAH-64) on each side for carrying Hellfire missiles, 2.75-inch folding fin aerial rockets, or external fuel tanks. The test aircraft differed from a production aircraft in several respects. It was fully instrumented for handling qualities, performance, and structural tests. No fire control computer (FCC) was installed. The backup bus controller (BBC) provides some of the functions of the FCC when the FCC is not operational. However, to use the full capability of the Digital Automatic Stabilization Equipment (DASE) the FCC must be operational on a production aircraft. For this evaluation, the software in the BBC was modified to provide full DASE capability. A test airspeed

boom mounted on the nose of the aircraft and a round instrumentation canister mounted on the top of the main rotor mast were items not found on production aircraft. Further description of the helicopter may be found in appendix B of this report, in the system specification (ref 1, app A), and in the operator's manual (ref 11).

TEST SCOPE

4. Flight testing was conducted in Mesa, Arizona (elevation 1387 feet) between 6 August and 29 August 1984. Twenty test flights were conducted for a total of 32.2 hours (22.1 hours productive). HHI installed, calibrated, and maintained the test instrumentation and performed all aircraft maintenance during the test. Flight restrictions contained in the Airworthiness Release issued by AVSCOM (ref 12) and the operator's manual were observed during this evaluation. Test conditions are presented in table 1. Where possible, flight test data were compared with the system specification and A&FC 3. Performance data were also compared to the data presented in the operator's manual.

TEST METHODOLOGY

5. Test methods are briefly discussed in the Results and Discussion section of this report. 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 program. A detailed listing of the test instrumentation is contained in appendix C. Test techniques and data analysis methods are described in appendix D. The Handling Qualities Rating Scale (HQRS) and the Vibration Rating Scale (VRS), shown in appendix D, were used to quantify pilot comments.

Table 1. Test Conditions[1]

Test	Calibrated Airspeed (knots)	Density Altitude (feet)	Gross Weight (lb)	Longitudinal Center of gravity (FS)	Remarks
Level flight Performance	41 to 139	7530	14,560	206.2	C_T[2] = 0.007991
		9620	14,490	206.6	C_T = 0.008488
		10,800	14,770	206.7	C_T = 0.008978
		12,920	14,650	206.2	C_T = 0.009525
		13,660	15,020	206.1	C_T = 0.010002
Control Positions in Trimmed Forward Flight	42 to 177	10,800			Level and Diving Flight
	59 to 133	7430			IRP[3] Climbs
	59 to 134	7300			Autorotational Descent
Static Longitudinal Stability	70	7140			Level Flight
	139	6880			Level Flight
	81	7440			IRP Climb
	81	7340			Autorotational Descent
Static Lateral-Directional Stability	70	7250			Level Flight
	139	7050			Level Flight
	81	7250			IRP Climb
	80	6630			Autorotational Descent
Maneuvering Stability	135	7170			Left and right turns pull ups and pushovers
Dynamic Stablility	70 and 139	7100	14,700	206.4 (AFT)	Longitudinal long-term Longitudinal, lateral, Directional pulses, DASE[4] ON, Attitude Hold ON and OFF DASE OFF
Power Management	0 to 130	6150 to 7290			Quick Stops Pull ups Recoveries from Autorotation
Instrument Flight Capability	0 to 140	2500 to 8000			Enroute Navigation Holding Approaches Simulated Emergencies
DASE Evaluation	0	3700			HAS[5] ON and OFF
	70 and 139	7100			Attitude Hold ON and OFF Level and Turning Flight
Simulated Single Engine Failures	0	2500			Hover
	70 100, 135	7450			Level and Climbing Flight at IRP
Vibration Characteristics	42 to 177	10,800			Level Flight and Dives

NOTES:

[1]8-Hellfire configuration, 100% rotor speed, DASE ON unless otherwise noted.
[2]C_T = Thrust coefficient.
[3]IRP = Intermediate Rated Power.
[4]DASE = Digital Automatic Stabilization Equipment.
[5]HAS = Hover Augmentation System.

RESULTS AND DISCUSSION

GENERAL

6. Level flight performance and handling qualities tests were conducted to verify that the production aircraft characteristics are the same as those of the prototype YAH-64 as tested in the A&FC 3 tests. The level flight performance and handling qualities of the production aircraft were not significantly different from those of the prototype. Additionally, the performance data presented in the operator's manual is a good representation of the production aircraft performance. One enhancing characteristic was found (the automatic contingency power feature of the T700-GE-701 engine). Two deficiencies were also found (the false indication of engine failures which was previously reported and the poor engine/airframe response characteristic which was previously reported as a shortcoming). Four shortcomings of the prototype aircraft have been corrected in the production aircraft while 13 remain. Additionally, 13 new shortcomings were found.

LEVEL FLIGHT PERFORMANCE

7. Level flight performance tests were conducted at the conditions of table 1 to verify that the performance data in the AH-64A operator's manual (which was based on ref 9 testing) accurately represents the production aircraft performance. The aircraft was flown at zero sideslip and a main rotor speed of 290 RPM. All level flight performance tests were conducted with a dummy 30mm chain gun in the 11 degree elevation, and 0 degree azimuth position without the ammunition chute installed. TADS and PNVS sights were in the stowed position. The Hellfire missile launchers were set at the zero degree elevation position. The environmental control subsystem (ECS) was operated at a level for pilot and copilot comfort and the stabilator was in the automatic mode. The aircraft was flown in the 8-Hellfire configuration at an aft longitudinal center of gravity (cg). Previous data were gathered at a forward cg which is a more adverse condition for level flight performance. During this test, the aircraft could not be flown at a forward cg because of a fuel quantity restriction for the forward fuel cell. Test results are presented nondimensionally in figures 1 through 3, appendix E. Dimensional test results are presented in figures 4 through 9. A list of external items, either installed on the test aircraft and not included in the system specification (ref 1, app A), or defined in reference 1 and not on the test aircraft during this evaluation is presented in table 1, appendix B. No corrections for electrical load, variable power consumed by the ECS or external configuration differences between the test and production aircraft were applied to the data in this report.

8. The maximum cruise airspeed using maximum continuous power with the aircraft in the 8-Hellfire configuration at the primary mission gross weight of 14,694 pounds and at 4000 feet pressure altitude with an ambient temperature of 35°C and 289 rotor RPM (100 percent) was determined to be 144 knots true airspeed (KTAS). During A&FC 3, it was determined that a change in cg from fuselage station (FS) 202 to FS 206 caused a drag decrease of 3.1 ft^2, equivalent flat plate area (based on engine shaft horsepower change). Applying the correction to the A&FC 3 data, the maximum speed is 145 KTAS, which agrees with the current results (see fig. 9, app E). Comparisons between the current results and the operator's manual also indicated good agreement. The level flight cruise performance of the AH-64A is not significantly different from that of the YAH-64 and the operator's manual cruise data accurately represents AH-64A level flight performance.

HANDLING QUALITIES

General

9. AH-64A handling qualities were evaluated with the production DASE software program No. 7-211D00005-11 (-11 version) except where noted. Additional testing included an evaluation of instrument flight capability in light turbulence and a night evaluation. System specification requirements were evaluated, where applicable. Both quantitative data and qualitative pilot comments were recorded during these tests. Tests were conducted in the 8-Hellfire configuration at the conditions specified in table 1.

Control System Characteristics

10. The control system mechanical characteristics were evaluated on the ground with external hydraulic and electric power applied to the aircraft and the rotors stopped. All measurements were taken at the pilot station with a digital force gauge and onboard control position indicators. Tests were performed with the trim feel system ON and OFF. The longitudinal control system was evaluated with the copilot/gunner (CPG) cyclic stick extended and retracted which affected the mass balance. Results were quantitatively verified with the auxiliary power unit ON (rotors static) and qualitatively verified in flight. Data are presented in figures 10 through 14, appendix E and a summary of control system mechanical characteristics is presented in table 2. The control system characteristics in the longitudinal and lateral axes appeared to be degraded compared to previous evaluations (refs 5 and 8, app A), due to the increased breakout plus friction forces

Table 2. Control System Mechanical Characteristics[1]

Test Parameter	Longitudinal (CPG[2] Stick Retracted)	Longitudinal (CPG[2] Stick Extended)	Lateral	Directional
Breakout Force (Plus Friction) (lb)	2.6 FWD, 3.4 AFT	3.0 FWD, 3.0 AFT	2.5 left, 2.0 right	9.0 left, 6.0 right
Full Control Travel (in.)	10.5	10.5	8.45	6.06
Control Oscillation	None	None	None	None
Freeplay	Negligible	Negligible	Negligible	Negligible
Force to Move Control 0.5 in. from Trim (lb)	3.3 FWD, 3.95 AFT	3.8 FWD, 3.4 AFT	2.9 left, 2.3 right	9.5 left, 7.5 right
Limit Control Force (lb)	17.0 FWD, 18.0 AFT	11.5 FWD, 17.6 AFT	9.1 left, 8.3 right	N/A
Control Centering	Positive	Positive	Positive	Positive
Trim Control Displacement Band (in.)	0.3	0.4	0.3	1.0
Control Jump	Negligible	Negligible	Negligible	Negligible
Control Forces Trimmable to Zero	No	Yes	Yes	Yes
Force Gradient (lb/in.)	1.45 FWD, 1.54 AFT	1.35 FWD, 1.45 AFT	0.79 left, 0.68 right	1.7 left, 2.6 right[3]
Average Friction Band (lb)	1.5 FWD, 0.8 AFT	1.5 FWD, 0.5 AFT	2.6 left, 2.0 right	10.6 left, 10.2 right

NOTES:

[1]Rotors static, primary hydraulic system and electrical power applied.
[2]Copilot gunner cyclic control stick.
[3]Force gradient increases sharply during last 3/4" of directional movement in both directions.

and the large mass imbalance in the longitudinal axis with the CPG cyclic stick retracted.

11. Longitudinal forces were measured at the lower third of the cyclic grip. With trim feel ON the control centering was positive with a small trim control displacement band (0.3 in.). In flight with the CPG cyclic stick retracted (normal mission configuration) the longitudinal control system showed stick mass imbalance in that slight aft pressure on the cyclic was required to maintain the desired trim attitude (para 28). On the ground with force trim OFF and the cyclic stick positioned 3.5 in. or less from the forward stop, the cyclic would fall forward. While airborne with force trim OFF the cyclic stick would rapidly fall toward the forward stop regardless of initial position as indicated in figure 15, appendix E. With the CPG cyclic stick extended mass imbalance was negligible. The longitudinal breakout force was high and demonstrated asymmetry (2.6 lb forward, 3.4 lb aft) with the CPG stick retracted. Although the asymmetry was not markedly objectionable to the pilot, the high cyclic control forces required constant retrimming during maneuvering flight. There was no mechanical coupling. The longitudinal control system characteristics failed to meet the following requirements of reference 1, appendix A.

a. 10.3.2.1.2 - the longitudinal breakout force (plus friction) in the fore and aft direction with the CPG cyclic stick retracted exceeded the maximum 10% asymmetry by 3%.

b. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the aft direction with the CPG cyclic stick retracted exceeded the 2.4 lb limit by 1.0 lb.

c. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the forward and aft directions with the CPG cyclic stick extended exceeded the 2.4 lb limit by 0.6 lb.

d. 10.3.2.5 - the longitudinal control force could not be trimmed to zero in flight with the CPG cyclic stick retracted.

12. The lateral control system forces were measured at the lower third of the cyclic grip. The lateral control system exhibited positive centering and a small (0.3 in.) trim control displacement band. The breakout force (plus friction) was higher left (2.5 lb) than right (2.0 lb) but the asymmetry was not noticeable to the pilot. However, during maneuvering flight and landing approaches, constant retrimming was required by the pilot to reduce control forces. Trim feel OFF operation was characterized by no force gradients or control centering. In the lateral

axis, the cyclic stick did not exhibit mass imbalance. The cyclic control system demonstrated force gradient harmony between the longitudinal and lateral axes. The lateral control breakout force (plus friction) failed the requirements of reference 1, appendix A, para 10.3.2.1.3 in that the 1.5 lb limit was exceeded by 1.0 lb (left) and 0.5 lb (right).

13. The directional control system characteristics were asymmetrical but not objectionable to the pilot. Control centering was positive but a large, noticeable trim displacement band existed (0.7 in.). The force gradient was approximately linear in both directions except at the last 3/4 in. where the gradient increased substantially due to the control tubes contacting guide sleeves in the tail boom area. No undesirable control oscillations or control jump was noted in this axis. The directional control mechanical characteristics failed the following requirements of reference 1, appendix A.

a. 10.3.2.1.3 - the left directional breakout force (plus friction) exceeded the 7.0 lb limit by 2.0 lb.

b. 10.3.2.2 - the left control force versus displacement gradient was less than the minimum value of 2.0 lb/in. by 0.3 lb/in.

c. 10.3.2.2.1 - the directional force gradient was not linear throughout the limit of control travel and was concave upward instead of downward during the last 3/4 in. of control travel in each direction.

14. Collective forces were measured at the forward edge of the grip, approximately one inch behind the throttle chop control. The average breakout (plus friction) was 3.3 lb down and 2.6 lb up at the mid range of collective movement. However, the friction was not constant throughout the range of collective control travel and an occasional racheting effect reduced force as much as one pound during movement. The poor design of the collective pitch control friction mechanism is a shortcoming previously reported. During maneuvering flight with positive g loading, the collective control consistently would drop up to one inch with friction set at mid level (5) or lower. The collective control system failed the requirements of reference 1, para 10.3.2.1.3 in that the collective breakout (plus friction) in the up direction was lower than the minimum value of 3.0 lb by 0.4 lb.

15. The control characteristics of the power control levers (PCL) were evaluated at the upper grip of each power lever. The breakout force (plus friction) of each engine control is summarized in table 3.

Table 3. Engine Control Breakout Forces[1]

Test Parameter		Breakout (Plus Friction) (lb)	
PCL[2]	Position	Forward	Aft
#1	Idle	6.0	6.0
	Mid	6.0	8.5
	Fly	7.5	5.0
#2	Idle	6.0	7.0
	Mid	9.0	10.0
	Fly	8.0	7.0

NOTES:

[1]Friction OFF
[2]Power Control Lever (PCL)

With the engine control friction ON, the average breakout plus friction in either direction was approximately 17 lb for the #1 PCL and 18 lb for the #2 PCL. Precise engine power lever movement was difficult even with friction OFF and remains a shortcoming previously reported. The excessive control force required to operate the engine power control levers failed to meet the requirements of para 10.3.3.2.3 of reference 1 by up to 2.5 lb (friction OFF) and 10.5 lb (friction ON).

Control Positions in Trimmed Forward Flight

16. Control positions in forward flight were evaluated in level flight, intermediate rated power (IRP) climb, and autorotative descent at the conditions presented in table 1. Data are presented in figures 16 and 17, appendix E. No variation of longitudinal control position with calibrated airspeed was found between 40 and 60 knots calibrated airspeed (KCAS) in level flight. Above 80 KCAS the gradient increased to 0.03 in./knot up to maximum level flight airspeed at intermediate rated power (V_H). The variation of lateral control position throughout the envelope was less than 1/4 in. Pitch attitude variation was essentially linear with increasing nose-down attitude with increased airspeed. In IRP climbs longitudinal control position was constant between 60 and 87 KCAS. Over 87 KCAS the control position moved forward with increased airspeed. In descent, longitudinal control position was constant between 60 and 75 KCAS, moved forward between 75 and 100 KCAS, and was approximately constant above 100 KCAS. The variation of lateral and directional control position was minimal throughout the airspeed range tested. Pitch attitude decreased slightly with airspeed in climbs and remained approximately constant in descents. Adequate control margins were available

throughout the conditions tested. The control positions in trimmed forward flight are satisfactory and met the requirements of paragraphs 10.3.3.1.2, 10.3.4.1.1, and 10.3.5.2.4 of the system specification (ref 1, app A).

Static Longitudinal Stability

17. The static longitudinal stability was evaluated in level flight, IRP climbs, and autorotative descents at the conditions presented in table 1. The collective was held fixed while airspeed was varied 20 knots in 5-knot increments about trim with the cyclic only. Data are presented in figures 18 through 21, appendix E. For the low-speed trim point, approximately 70 KCAS, the longitudinal control position gradient was essentially neutral within 12 knots from trim varying less than 0.1 in. At high speed (139 KCAS) the stick-fixed longitudinal stability was weakly positive as indicated by the shallow control position gradient. At both airspeeds, stick-free static longitudinal stability was neutral within 10 knots of trim and slightly positive outside this range as indicated by the qualitative variation of longitudinal control force with airspeed (increasing forward force with increased airspeed). In IRP climbs at 81 KCAS, the longitudinal control position gradient was essentially neutral within 15 knots of trim airspeed. In descents at 81 KCAS the gradient was positive within 10 knots of trim airspeed. Although the neutral to weak longitudinal stability contributed to poor trimmability and increased pilot workload in simulated instrument meteorological conditions (IMC), as mentioned in paragraph 28 and in previous evaluations, it is satisfactory for attack helicopters operating in visual conditions. The static longitudinal stability failed to meet the requirements of paragraph 10.3.4.1 of the system specification (ref 1), in that the variation of longitudinal control position with airspeed was neutral in level flight at a trim airspeed of 70 KCAS and in IRP climb at 81 KCAS.

Static Lateral-Directional Stability

18. Static lateral-directional stability was evaluated in level flight, IRP climbs, and descents at the conditions presented in table 1. The collective control was held constant and sideslip angle was varied in 5-degree increments (left and right) while maintaining constant airspeed and heading. Data are presented in figures 22 through 25, appendix E. At all conditions tested the aircraft exhibited positive directional stability (as indicated by increased left directional control with increased right sideslip). Positive dihedral effect (as indicated by increased right lateral control with increased right sideslip) was also exhibited at all conditions. Sideforce characteristics were weak

at 70 KCAS in level flight (approximately 0.14 deg roll attitude per deg of sideslip). At 139 KCAS in level flight sideforce characteristics were noticeably higher (approximately 1.0 deg roll attitude per deg of sideslip). The static lateral-directional stability is satisfactory and met the requirements of paragraphs 10.3.5.1.5, 10.3.5.1.6, and 10.3.5.1.7 of the system specification (ref 1, app A).

Maneuvering Stability

19. Maneuvering stability was evaluated in constant airspeed ball-centered turns and in pull ups and pushovers at the conditions specified in table 1. Data are presented in figure 26, appendix E. Maneuvering stability, as indicated by the variation of longitudinal control position and force with normal acceleration was positive (aft control movement and increasing aft force with increasing load factor) at 135 KCAS. The helicopter attitude was predictable and load factor could be quickly and easily commanded by the pilot. However, in constant airspeed steady turns of more than 60 deg of roll attitude, airspeed ($\pm$10 kt), bank angle ($\pm$5 deg), and pitch attitude ($\pm$10 deg) were difficult to maintain in that constant control inputs in all axes ($\pm$1/2 in) were required. During pullups at the never exceed airspeed with IRP the aircraft tended to "dig in" even with small longitudinal inputs. With a small power reduction this characteristic was eliminated. The maneuvering stability is satisfactory and met the requirements of paragraph 10.3.6 of the system specification (ref 1, app A) with the exception of 10.3.6.2 which was not evaluated.

Dynamic Stability

20. The longitudinal long-term dynamic stability was evaluated with the test DASE hardover computer installed (-11 software) at the conditions in table 1. Aircraft motion was induced by displacing longitudinal cyclic from trim and increasing or decreasing airspeed by 10 knots indicated airspeed (KIAS) then returning the cyclic slowly to trim. All controls were then held fixed until recovery was initiated. Tests were conducted with DASE ON (including attitude hold ON and OFF) and DASE OFF. Time history data are presented in figures 27 through 32, appendix E. With attitude hold ON, the response to an airspeed displacement was essentially deadbeat with the aircraft returning to the trim airspeed and attitude. With DASE ON and attitude hold OFF, the long-term response was dynamically unstable with a period of approximately 67 seconds at a trim airspeed of 60 KIAS (boom) and 65 seconds at 126 KIAS (boom) (V_H). The amplitude increased moderately after the first cycle and approaching attitude/airspeed

limits required the pilot to initiate recovery after one cycle at 60 KIAS (boom) and two cycles at V_H. With attitude hold OFF and the CPG stick retracted, the long-term response was easily excited due to the combination of weak static stability, stick mass imbalance, and the lack of absolute positive centering (fig. 32). While this characteristic is undesirable and annoying to the pilot, it was not a factor during maneuvering flight and was only noticeable in cruise. The use of attitude hold prevented attitude/airspeed excursions during the enroute phase of flight. With the DASE OFF, the long-term oscillation amplitude increased rapidly at both airspeeds requiring recovery within one cycle due to rapid pitch rates (up to 20 deg/sec) and high g onset. The recovery to controlled flight was accomplished easily in any DASE configuration. However, with DASE OFF at V_H, load factors up to 2.5 g and high pitch rates were experienced, which required timely and positive control inputs by the pilot to prevent exceeding aircraft limits. The DASE ON long-term response is satisfactory. The DASE OFF long-term response is satisfactory for a degraded mode. The long-term response met the requirements of paragraph 10.3.4.2 of the system specification (ref 1, app A).

21. The short-term longitudinal and lateral-directional dynamic stability were evaluated at the conditions presented in table 1. Aircraft motion was induced by 1 inch 1/2 second pulses in all axes. The controls were then held fixed until the motion subsided or recovery was initiated. Time history data are presented in figures 33 through 40, appendix E. With attitude hold ON or OFF and DASE ON or OFF, the longitudinal short-term response was essentially deadbeat. The lateral-directional response was dynamically stable and damped out within one cycle with DASE ON (attitude hold ON or OFF). DASE OFF, the lateral-directional oscillations were neutrally damped but controllable. The short-term longitudinal and lateral-directional dynamic stability met the requirements of paragraphs 10.3.4.2 and 10.3.5.3 of the system specification (ref 1, app A).

Power Management

22. The power management features were evaluated at the conditions shown in table 1. Tests included engine/airframe response characteristics, functional checks, and the contingency power capability. The functional checks and contingency power capability remained unchanged from A&FC 3 and are satisfactory and enhancing, respectively. The undesirable engine/airframe response is essentially unchanged from A&FC 3, however, due to the additional poor engine/rotor matching characteristics noted during pull up maneuvers the previously noted shortcoming has been downgraded to a deficiency.

23. The engine/airframe response tests included power recoveries from autorotation, collective control reversals, and various mission representative maneuvers (quick stops, rapid lateral control inputs, and pull ups). Time history data are presented in figures 41 through 47, appendix E. Following collective applications from below 6% torque during recoveries from autorotations and quickstops, the most common response noted was a main rotor speed droop accompanied by activation of the low main rotor speed warning and subsequent engine/airframe oscillations (figs. 41 and 42). Main rotor speed varied from 94 to 104 percent and oscillations damped in approximately 12 seconds over four cycles. Significant engine/airframe response was also noted during the recovery from pull ups from below one g (symmetrical pull ups). When initiating this maneuver with the collective fixed at above 90% gas producer speed (N_G) (approximately 40% torque), no objectionable adverse engine/airframe response was noted (figs. 43 and 44). When initiating the same maneuver from below this initial power condition (40% torque), main rotor speed droop and activation of the low main rotor warning was observed accompanied by engine/airframe oscillations (figs. 45 and 46). The oscillations damped in approximately 15 seconds and 3 cycles. Main rotor speed varied from 90 to 105 percent (maximum transient limit 104 percent) and yaw rates up to 8 deg/sec in each direction were noted. Initial trim airspeed prior to the pull up maneuver did not appear to be a major factor in the subsequent response. Any down collective movement by the pilot during the pull up greatly increased main rotor overspeed and during the recovery phase significant droop generally occurred. Figure 47 is a representative time history of a 2.2 g pull up initiated at 120 KIAS (boom) and approximately 35% torque. During the pull up, the collective was unintentionally lowered 1.8 in. from the initial trim position of 5.1 in. The subsequent main rotor speed reached 114% during g onset and then drooped to 88% accompanied by the low main rotor speed warning and both ENG OUT warning lights (para 36). Maintaining simultaneous heading ($\pm$5 deg), roll attitude ($\pm$5 deg), and rotor speed (94 to 104%) tolerances required in the NOE environment were not possible even with large control inputs in all axes (HQRS 7). The engine/airframe responses observed during this test will be most distracting during NOE or contour flight while performing recoveries from low power descents, quickstops, and pull up type evasive maneuvers where ground clearance and power required are critical. The pilot will be required to redirect his attention inside the cockpit to compensate for the main rotor speed fluctuations and airframe oscillations, resulting in the degradation of the masking and maneuvering capability of the aircraft. This problem will severely reduce mission effectiveness and may result in the helicopter unintentionally striking the ground (with loss of

aircraft and crew) during rapid descent recoveries when threat avoidance or combat maneuvering tactics are employed. The undesirable engine/airframe response during power applications from a zero torque condition and during pull ups from below 40% torque at less than one g is a deficiency which should be corrected prior to operational use of the helicopter.

24. During the evaluation of mission type maneuvers, it was noted that rapid left lateral inputs larger than 2 in. produced large torque spikes (more than 30% above trim power) (fig. 48, app E). This torque spike was more pronounced at high forward airspeeds (near V_H) at high power settings. The dual engine torque transient limit is 115% and could easily be exceeded during evasive left rolling maneuvers while operating above 85% torque. The following CAUTION should be placed in the operator's manual.

CAUTION

> Rapid left roll inputs greater than 2 inches could exceed dual engine transient torque limits when operating above a trimmed torque setting of more than 85%.

Instrument Flight Capability

25. A limited IMC evaluation was performed in daytime with light turbulence and at night with calm air conditions. Turbulence reporting criteria are defined in table 1, appendix D. Maneuvers performed included climbs, descents, climbing and descending turns, changing airspeed, retrimming in level flight and turns, simulated instrument approaches and instrument takeoffs. IMC conditions were simulated using a helmet mounted hood. The pilot's Visual Display Unit (VDU) (fig. A) was used as the primary flight reference to evaluate the symbology available to the pilot through the Integrated Helmet and Display Sighting System. The VDU provided pitch and roll attitude, vertical speed indicator (VSI) with digital altitude readout below 1500 ft above ground level, aircraft heading, torque, and true airspeed information. Aircraft flight and navigation indicators were used to provide other information as required and to confirm VDU information. The skid/slip ball on the VDU display was inoperative. The pitch attitude reference was not incremented and provided only trend information. Additionally, the VSI and VDU display showed rate of climb and descent approximately 50% greater than the pilot's VSI. The automatic direction finding (ADF) navigation system failed to give reliable indications when the aircraft was more than 3 miles from the station. Furthermore, the aural

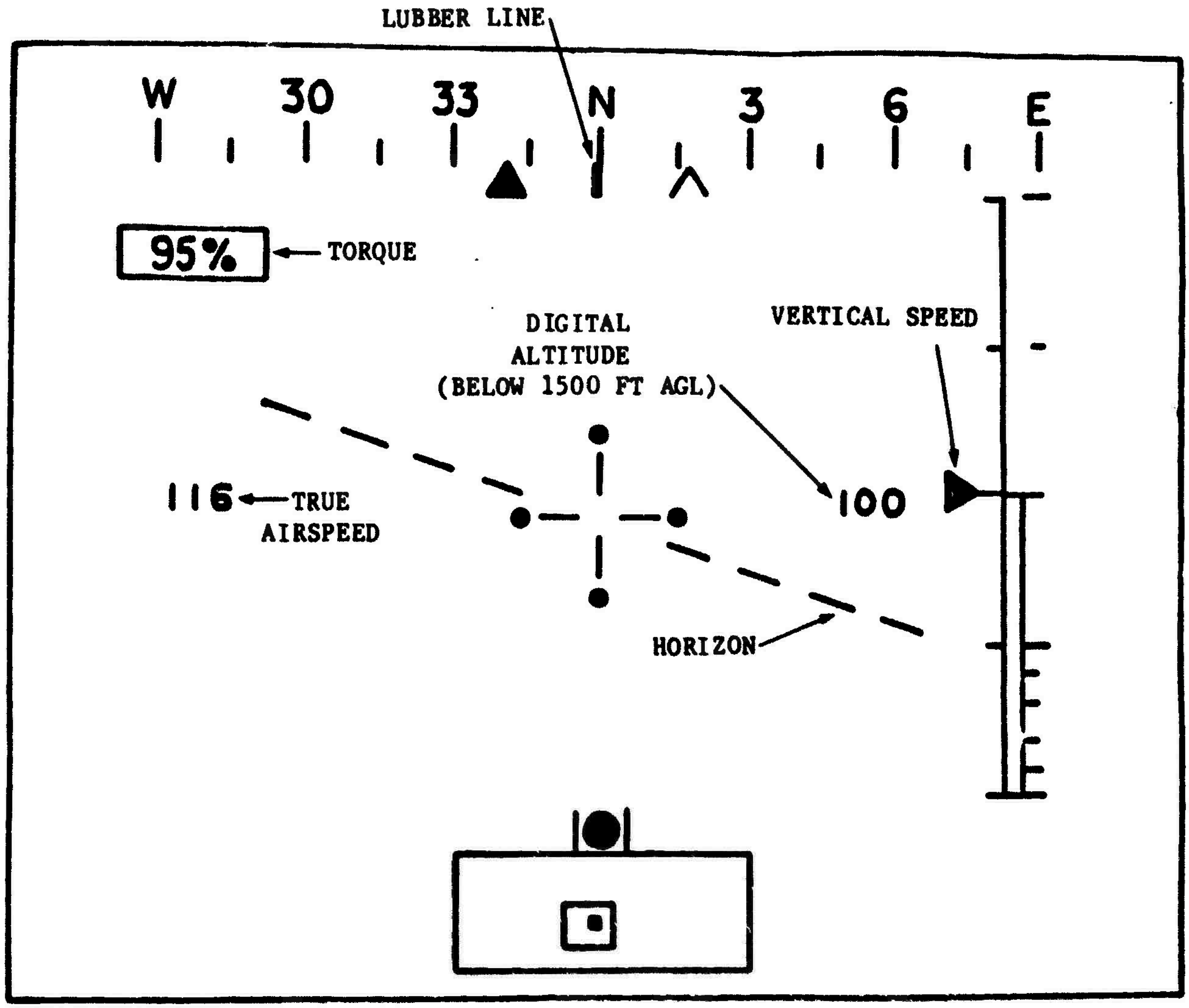


Figure A. Visual Display Unit (VDU)

identification on the ADF receiver was unreadable due to loud static in the background. The lack of reliable audio signal and azimuth information from the AN/ARN 89B ADF set is a shortcoming.

26. Instrument approaches were executed using the Precision Approach Radar at Luke Air Force Base, Arizona, Auxiliary Field Number One and using the Doppler to simulate automatic direction finding stations. The PNVS system was not operational in the test aircraft. Maneuvers were flown DASE ON with attitude hold ON and OFF and with DASE OFF.

27. Flight with attitude hold ON, in light turbulence required minimal pilot inputs to maintain aircraft pitch attitude within ±3 degrees of trim in light turbulence (HQRS 3). Retrimming in level flight and turns (attitude hold ON) caused the stability augmentation system (SAS) actuators to recenter resulting in uncommanded pitch and roll inputs which required the pilot to apply corrective inputs in all three axes. Several large longitudinal inputs of 1 to 1-1/2 inch were required to obtain the desired pitch attitude and airspeed (HQRS 4) due to the recentering of the SAS actuator. The pilot tended to fly against the force gradients rather than retrim when small airspeed changes were made. Pilot workload increased significantly from calm air to light turbulence due to the increased requirement to retrim the aircraft (particularly in pitch). The excessively large longitudinal cyclic inputs required when retrimming, during IMC flight with attitude hold ON remain a shortcoming (previously reported in A&FC 3).

28. Flight with DASE ON and attitude hold OFF in light turbulence increased the pilot workload significantly requiring frequent retrimming to maintain ±5 KIAS, ±10 degrees heading, and ±100 ft altitude (HQRS 5). Weak longitudinal static stability (para 17), trim control displacement band (para 11), and longitudinal stick mass imbalance (CPG cyclic retracted) (para 11) resulted in poor trimmability. The cyclic tended to creep forward requiring the pilot to position the cyclic aft of the desired position to compensate. Extending the CPG cyclic to the full up position reduced this problem. The IMC flight characterisitics with DASE ON and attitude hold OFF are satisfactory since this is a degraded mode of operation. The trim system failed to meet the requirements of paragraph 10.3.2.5 of the specification (ref 1, app A) in that cyclic control would not maintain the zero force position selected by the pilot. A comment should be added to the operator's manual stating "Intentional flight in IMC with attitude hold OFF is not recommended".

29. DASE OFF flight in light turbulence resulted in an extremely high pilot workload to maintain ±10 KIAS due to the divergent long-term characteristics (para 20). Continuous cyclic, collective, and pedal inputs were required to execute instrument approaches within safe limits. The IMC flight characteristics with DASE OFF are satisfactory for a degraded mode of operation. DASE OFF flight met the requirements of paragraph 10.3.2.7.3 of the specification (ref 1). A change should be made to the operator's manual to prohibit intentional flight in IMC with DASE OFF.

DASE Evaluation

30. Tests were conducted to evaluate the production DASE software, (-11 version). The production DASE configuration is described in appendix B. Gust response was evaluated in forward flight with attitude hold ON and OFF and at a hover with the Hover Augmentation System (HAS) ON and OFF. The aircraft gust response in forward flight and hover is essentially unchanged from that reported in A&FC 3 and remains satisfactory.

31. Tests were performed to investigate aircraft response to SAS actuator recentering in three axes (simulated DASE failure). Additionally, single axis SAS actuator hardovers were evaluated using a production DASE modified to include a hardover box. Test conditions are shown in table 1.

32. Simulated three axis DASE failures were performed during forward flight (attitude hold ON and OFF) and hover (HAS ON and OFF) by depressing the DASE release switch on the pilot cyclic grip. The aircraft was stabilized in trim prior to DASE disengagement (SAS actuators recentering). All controls were held fixed for a minimum of 3 seconds following disengagement. The most abrupt response in forward flight occurred during disengagement with attitude hold ON. The aircraft was easily returned to stable level flight following a 3 second delay. The most abrupt response during disengagement at a hover was with HAS ON. The aircraft was easily returned to a stable hover following a 3 second delay. The aircraft response to simulated DASE failures in forward flight and hover is satisfactory. The aircraft response to abrupt disengagement of the automatic stabilization equipment in forward flight and hover met the requirements of paragraph 10.3.2.7.1 of the specification (ref 1, app A).

33. Single axis DASE hardover conditions were evaluated during forward flight and hover using a production DASE modified to drive the SAS actuators in the pitch, roll, and yaw axes to full travel on command. Following the hardover condition, the DASE

disengaged only in the affected axis and the respective SAS actuator recentered. All controls were held fixed for a minimum of 3 seconds following the hardover condition. During forward flight and hover, the aircraft responded more abruptly to SAS actuator recentering than the initial hardover condition. The aircraft was easily returned to stable forward flight or hover following a 3 second delay. Aircraft response to single axis hardovers in the pitch, roll, and yaw axes is satisfactory.

Simulated Single Engine Failures

34. Simulated single engine failures were evaluated by retarding one power lever to the idle position. Test conditions are shown in table 1. Collective control reduction was initiated after 2.0 seconds delay time or 1.0 second after activation of the low rotor speed warning light and audio tone for specification compliance. The low rotor speed warning system was designed to activate at 94%. During the test, the low rotor speed system activated at 96 to 97% rotor speed (para 37). A representative time history is presented in figure 49, appendix E.

35. Initial aircraft response to a simulated single engine failure was a slight left yaw. Due to this mild aircraft response, the first indication of engine failure at high power settings may often be the activation of the engine out/low rotor speed warning audio tone. A collective control delay of 1.0 second after activation of the low main rotor speed warning resulted in a droop of the operating engine power turbine speed (N_P) to 89% during a simulated single engine failure at V_H. This was 5.0% below the minimum power off transient N_P limit of 94%. No adverse handling qualities were observed at main rotor speeds down to 85%. Main rotor speed increased from 85% to normal operating range (98 to 100%) within 4 seconds after initial collective reduction. Aircraft response to simulated single engine failure is satisfactory. The aircraft response to simulated engine failure failed to meet the requirements of paragraph 10.3.8.1.1 of the System Specification (ref 1, app A) in that the available collective control delay time at V_H was less than 2.0 seconds.

36. The engine out warning system was evaluated at high power settings by retarding one power lever to the idle position to determine if a false indication of a dual engine failure would occur. This was a previously reported deficiency during A&FC, part 3 (ref 9). The engine out warning system was designed to activate at 89% N_P speed with the power levers at the fly position. The engine out warning system was modified for this test so that the system was not deactivated when the power levers

were out of fly. At high power settings (N_G speeds above 95%), indications of a single engine failure were a low main rotor speed warning (horn and light) followed shortly by illumination of both engine out warning lights as main rotor speed and N_P of the operating engine drooped below 89%. This condition could result in the pilot executing the dual engine out emergency procedure during terrain flight operations and committing the aircraft to an autorotative landing when it may have been possible to establish single engine flight with the contingency power feature of the YT700-GE-701 engine. The possibility of a false indication of engine failure remains a deficiency. The engine out warning system should be modified to insure a timely warning of engine failure under all flight conditions and should not activate unless an actual engine failure has occurred.

37. The low main rotor speed warning system was evaluated to determine if the system activated at the design threshold of 94%. During this test, the low main rotor speed warning system consistently activated at 96 to 97% during simulated single engine failures. On several occasions the warning system activated at 99 to 100% rotor speed during normal flight maneuvers. Replacing the engine out warning box failed to correct activation of the warning system at excessively high rotor speed. Activation of the low main rotor speed warning system at excessively high rotor speed is a shortcoming.

VIBRATION

38. The forward flight vibration characteristics of the AH-64A were evaluated in the 8-Hellfire configuration. Quantitative data were recorded during level flight tests at the conditions shown in table 1. Quantitative vibration data are presented in figures 50 through 52, appendix E. Qualitative tests were conducted during climbs, descents, and low speed flight. Vibration assessment ratings were assigned in accordance with the VRS shown in figure 4, appendix D. Significant and objectionable 4 per revolution (4/rev) vibration accelerations (19.3 Hz) became noticeable to both crewmembers below 60 KCAS and above 110 KCAS (VRS 4). The 4/rev vibrations increased from VRS 4 to VRS 5 at 120 KCAS and VRS 6 above 140 KCAS. 4/rev vibrations were also noticeable during IRP climbs at 80 KCAS (VRS 5) and during normal approaches to a hover (VRS 5 below 35 KCAS). Vibrations observed during this evaluation were most noticeable at the pilots station as they have been during previous tests. The 4/rev vibration characteristics are essentially unchanged from those previously noted in part 3 of the A&FC report (ref 9, app A). The objectionable 4/rev vibration remains a shortcoming. Vertical vibration

characteristics failed to meet requirements of paragraph 3.2.11.5.1 of the System Specification (ref 1) in that 4/rev vibration intensities exceeded the specification limit below 80 KCAS and at 126 KCAS; the 8/rev vibration intensities exceeded 0.2g at 83 KCAS.

HUMAN FACTORS

Cockpit Evaluation

39. The pilot cockpit of the AH-64A was evaluated throughout the test program including an evaluation of cockpit design with primary emphasis on instrument displays, switch functions, control designs, and ease of operation of system controls. Pilot comfort was also considered. Uncorrected shortcomings previously reported are shown in paragraph 55.

40. Ease of system controls operation was evaluated during this test. Access to the pilot's #1 fire handle is partially obstructed by an instrument panel flood light mounted on the underside of the glare shield at the upper left side of the instrument panel. This flood light was removed from the test aircraft but is installed in current production aircraft. The position of the flood light will cause the pilot difficulty in reaching and pulling the #1 fire handle. Obstructed access to the pilot's #1 fire handle is a shortcoming.

41. When attempting fuel transfer or crossfeed operation (use fuel from one tank only) no immediate indication was available to show direction of transfer or that crossfeed operation was in progress. The primary means of monitoring fuel transfer or crossfeed operations is fuel quantity indications on the CPG digital indicator. The operator's manual (ref 11) indicates the fuel transfer light will be on during fuel transfer operations. Crew attention will easily be diverted from the fuel quanitity indicators by normal crew duties allowing more than the desired amount of fuel to be transferred/crossfed and could easily result in exceeding aircraft cg limits or engine flame out (fuel starvation) when operating in a low fuel condition. The lack of a fuel transfer/crossfeed light is a shortcoming. The fuel transfer light fails to meet the requirements of paragraph 3.7.7.1.3.10(C) (ref 1) in that the light does not illuminate when fuel transfer is in progress.

42. The pilot's cyclic grip (photo 1) incorporated eight different functions activated by thumb switches. In addition, some additional functions are required to be activated simultaneously such as the weapons action switch and the trigger switch. An

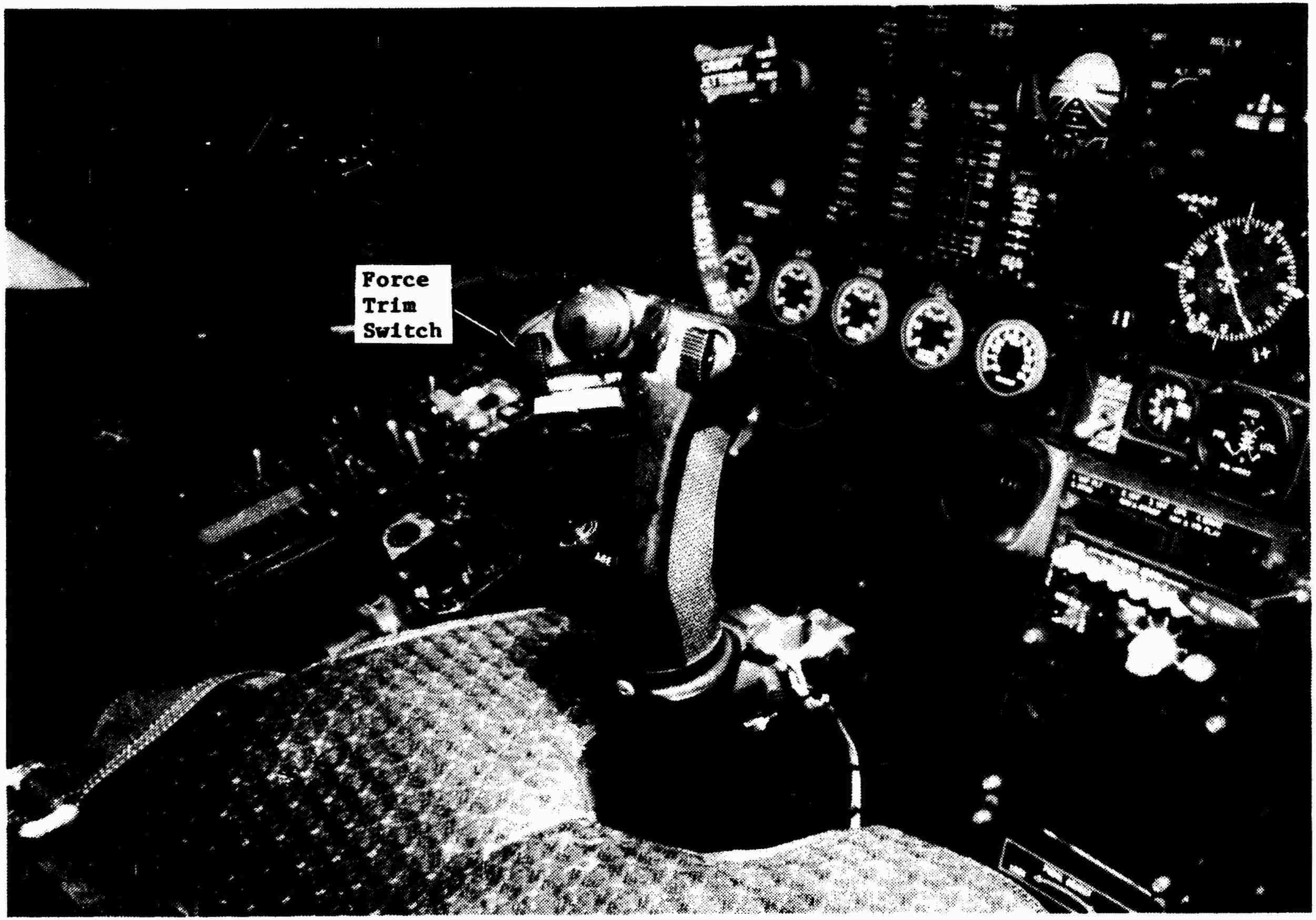


Photo 1. Pilots Cyclic Grip

extreme reach was required for operation of the force trim release switch. A pilot with an average sized hand (glove size 9) will have difficulty in operating the switches on the cyclic and will frequently be required to reposition his hand on the cyclic grip in order to perform the necessary functions. Frequent use of the force trim release switch was required during all flight and hovering maneuvers. Furthermore, the pilot was unable to utilize the force trim release switch and radio/intercom switch simultaneously. This severely limited internal and external communications during high gain tasks requiring control retrimming. The mission pilot will prefer to fly with force trim OFF during maneuvering flight and hover. Flight with force trim OFF is less than desirable due to the stick mass imbalance and loss of the Command Augmentation System (CAS). The poor anthropometric design of the pilot's cyclic grip remains a shortcoming.

43. The pilot fuel control panel (redesigned since A&FC, Part 3) shown in photo 2, was evaluated throughout the test. Switch labeling was not consistent between front and aft cockpit in that the "CROSSFEED" switch on the pilot's panel is labeled "TK SEL" on the CPG's panel (fig. 30, app B). Engine fuel switches were difficult to see with engine power control levers in the full aft position. Normal operation of the fuel system required frequent pilot attention to the fuel control panel in order to maintain comparable fuel levels in each tank. No immediate indications were available to show direction of transfer or that crossfeed operation was in progress (see para 41). Switch identification and determination of switch position were extremely difficult during night operations. The close proximity of the fuel transfer and crossfeed switches to the engine fuel switches and similar manner of switch operation will result in a high probability of inadvertent shutdown of an engine during attempted fuel crossfeed or transfer operation. The poor design of the pilot's fuel control panel remains a shortcoming.

44. The engine anti-ice system was qualitatively evaluated during this test. On two occasions an uncommanded open anti-ice/start bleed valve, with no positive cockpit indications, resulted in reduced power available and higher operating engine turbine gas temperature (TGT) at a given power setting. The anti-ice/start bleed valve and the engine inlet anti-ice valve are spring loaded open and electrically closed by the ENGINE INLET SWITCH on the pilot's anti-ice panel. ENG 1 and ENG 2 advisory lights on the anti-ice panel are designed to illuminate when the ENG INLET switch is placed on and the electrical circuit is complete. An anti-ice caution light for each engine is designed to illuminate when an engine nose gearbox fairing overheats (120 to 180°F). The lack of a positive indication of anti-ice/start bleed valve

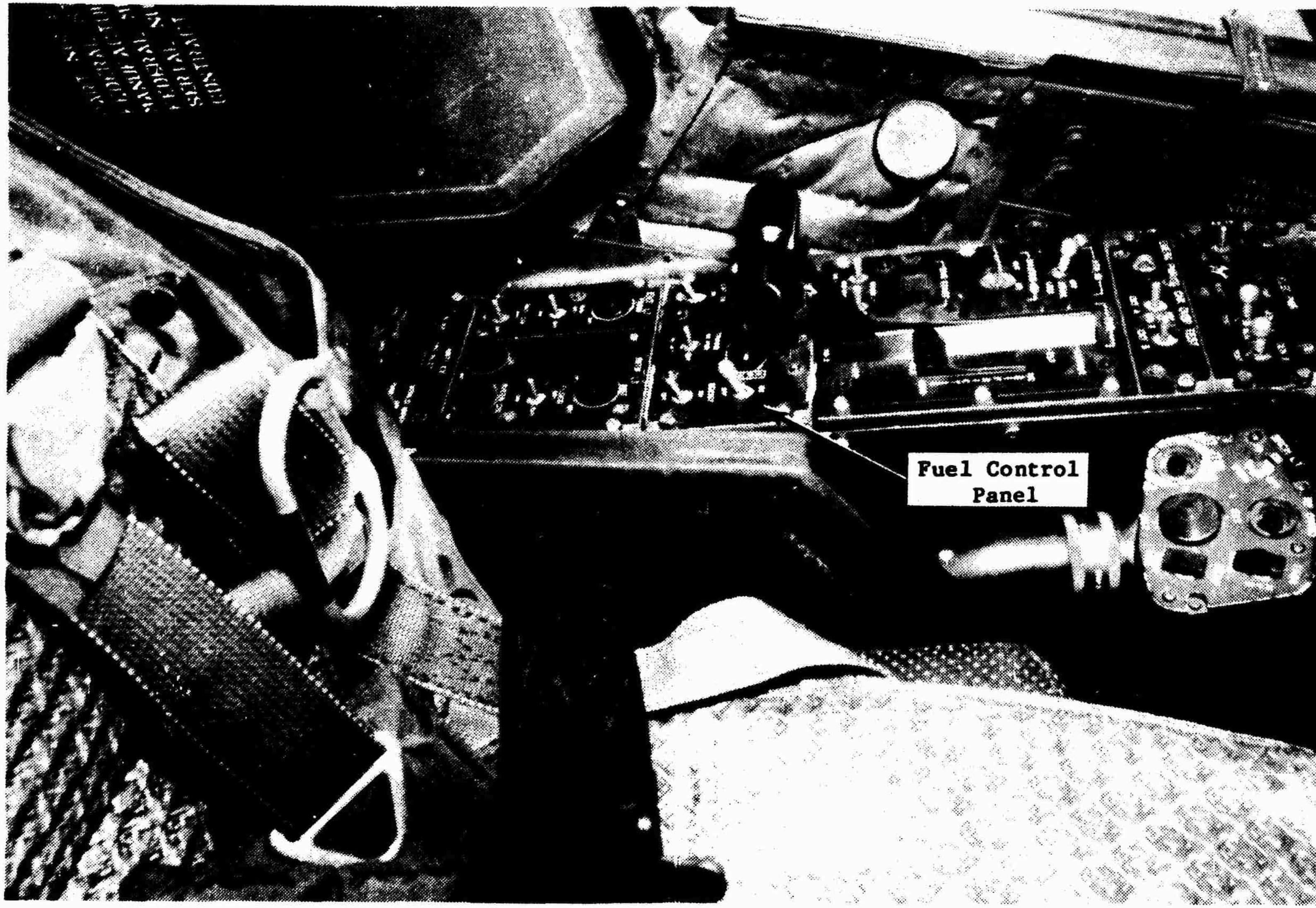


Photo 2. Pilots Fuel Control Panel

position is a shortcoming. The engine anti-ice system fails to meet the requirements of paragraph 3.7.5.4.1 of the System Specification (ref 1, app A) in that failure of the anti-ice/start bleed valve to close does not illuminate a caution light.

45. The VDU roll attitude reference was evaluated during night and daytime simulated instrument conditions. VDU symbology for roll attitude reference is a symmetrical dashed line (fig. A) which fails to identify upward from downward direction. During unusual attitude recovery from bank angles approaching 90 degrees, the pilot was unable to determine the appropriate direction of lateral control input required to recover the aircraft to wings level, upright flight using the VDU roll attitude reference. A standby attitude indicator is available on the instrument panel. However, the pilot will be using the integrated helmet and display sight system (IHADSS) display as the primary reference for aircraft attitude when inadvertent IMC conditions are encountered. There is a high probability of unusual aircraft attitudes during the initial entry into inadvertent IMC conditions. The lack of VDU roll attitude symbology to distinguish upward from downward direction is a shortcoming.

46. Failure of the ECS to provide adequate crew station cooling was a shortcoming previously reported in A&FC, Part 3 and resulted in a system redesign. The ECS also provides cooling air for the forward avionics bay (FAB). Mission equipment normally installed in the FAB was not operational on the test aircraft. The ECS light illuminated (indicating temperatures greater than 27°C in the FAB) during ground operations with outside ambient temperatures of 40°C and above. A deviation/change to normal operating procedures for production aircraft described in reference 13, appendix A, requires the crew to close the crew station outlets for 5 minutes or until the ECS light goes out to prevent damage to mission equipment in the FAB. This procedure was not implemented during the test due to the absence of mission equipment in the FAB. Crew station cooling air was adequate for the test aircraft. Further testing of a fully equipped mission aircraft should be accomplished to insure adequate cooling air is available for the crew station and FAB simultaneously.

Night Evaluation

47. The AH-64A was equipped with an An/APX-100(V) transponder and RT-1296 transponder control panel. The control head is mounted on the right hand console (photo 3) and requires the pilot to look down and to the right rear when setting or reading the transponder code numbers. The code numbers are displayed in small windows recessed below a cover plate which partially obscures

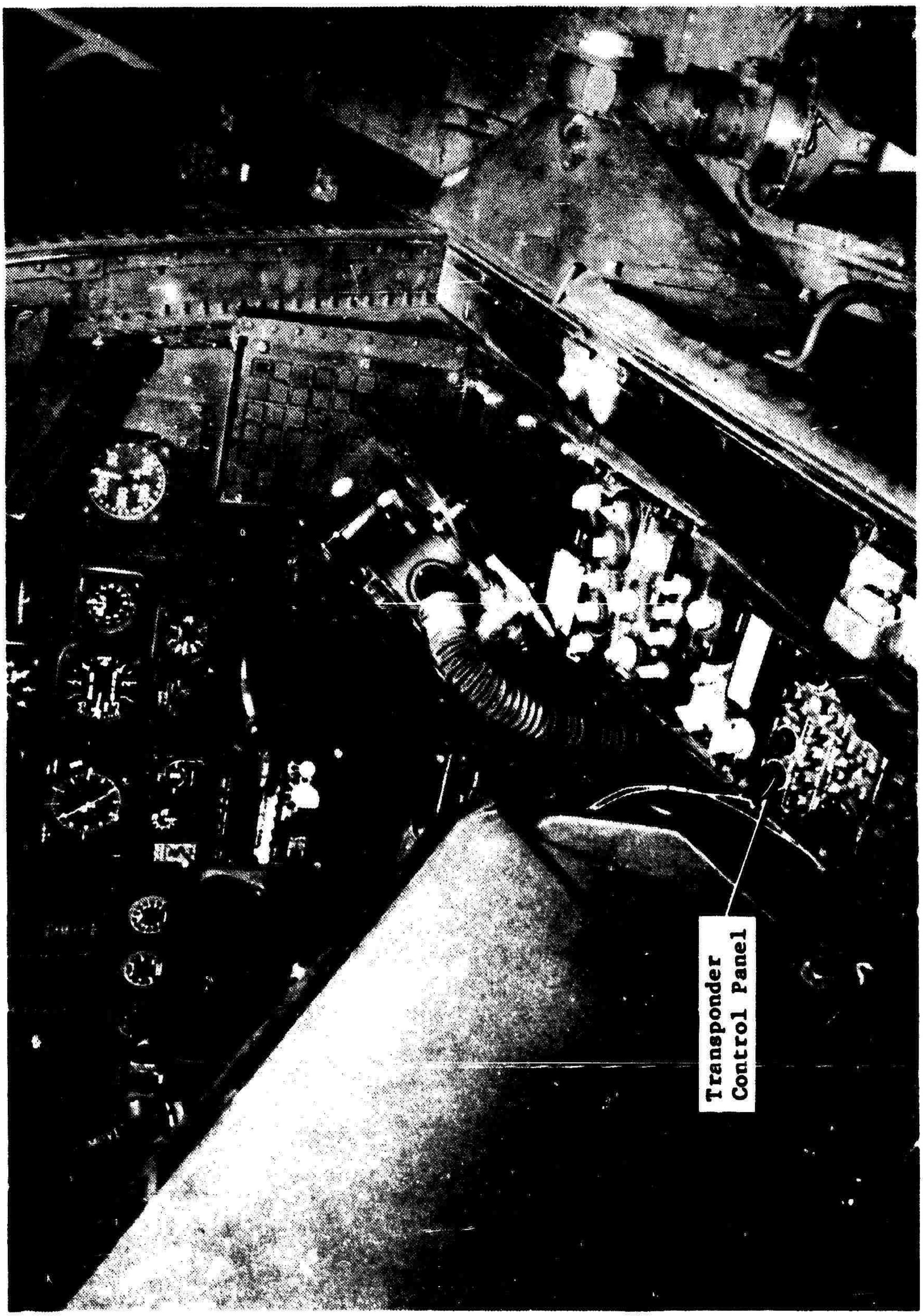


Photo 3. Pilots Right Console

the numbers being displayed. Additionally, a small amount of dust collected on the recessed windows which made the control numbers unreadable at night with available cockpit lighting. To adequately clean the code windows, the cover plate must be removed. In dusty conditions, the code windows must be cleaned daily. The partially obscured code numbers and dust collection on the code windows also made it difficult to read the numbers under daylight conditions. Poor readability of the transponder code numbers is a shortcoming.

48. Instrument panel and indicator lighting at both crew stations was evaluated during this test. Light intensity of both the pilot and CPG Marconi indicators is set with a single knob at the pilot station since the CPG does not have a separate light adjustment in the front cockpit. The lack of an individual light intensity adjustment for the CPG Marconi indicators is a shortcoming.

RELIABILITY AND MAINTAINABILITY

General

49. The reliability and maintainability features of the AH-64A aircraft were evaluated throughout the test. Seventeen Equipment Performance Reports (EPR's) (appendix F) were prepared and submitted during this test. This section is intended only to document undesirable reliability and maintainability features encountered during the test and not previously reported. Previously reported shortcomings are discussed in paragraph 55.

50. The Heading and Attitude Reference System (HARS) failed to align accurately during this test. Alignments, both the normal and fast mode, were inconsistent and indicated headings varied from 0 to 5 degrees from actual runway heading. Additionally, indicated heading errors varied from 0 to 9 degrees following flights of 1.0 to 1.5 hours. Attempts to realign the system (on the ground with rotors turning) using the 8 minute normal align feature were unsuccessful. The inconsistent and inaccurate alignment of HARS is a shortcoming.

Main Rotor Plunger

51. The main rotor system incorporated a plunger assembly (photo 4) designed to determine that the main rotor shaft nut and retention ring are installed. Presently, a daily special inspection procedure requires depression of the main rotor plunger. The reason for this inspection is not clear since the plunger does not indicate that there is torque on the retaining

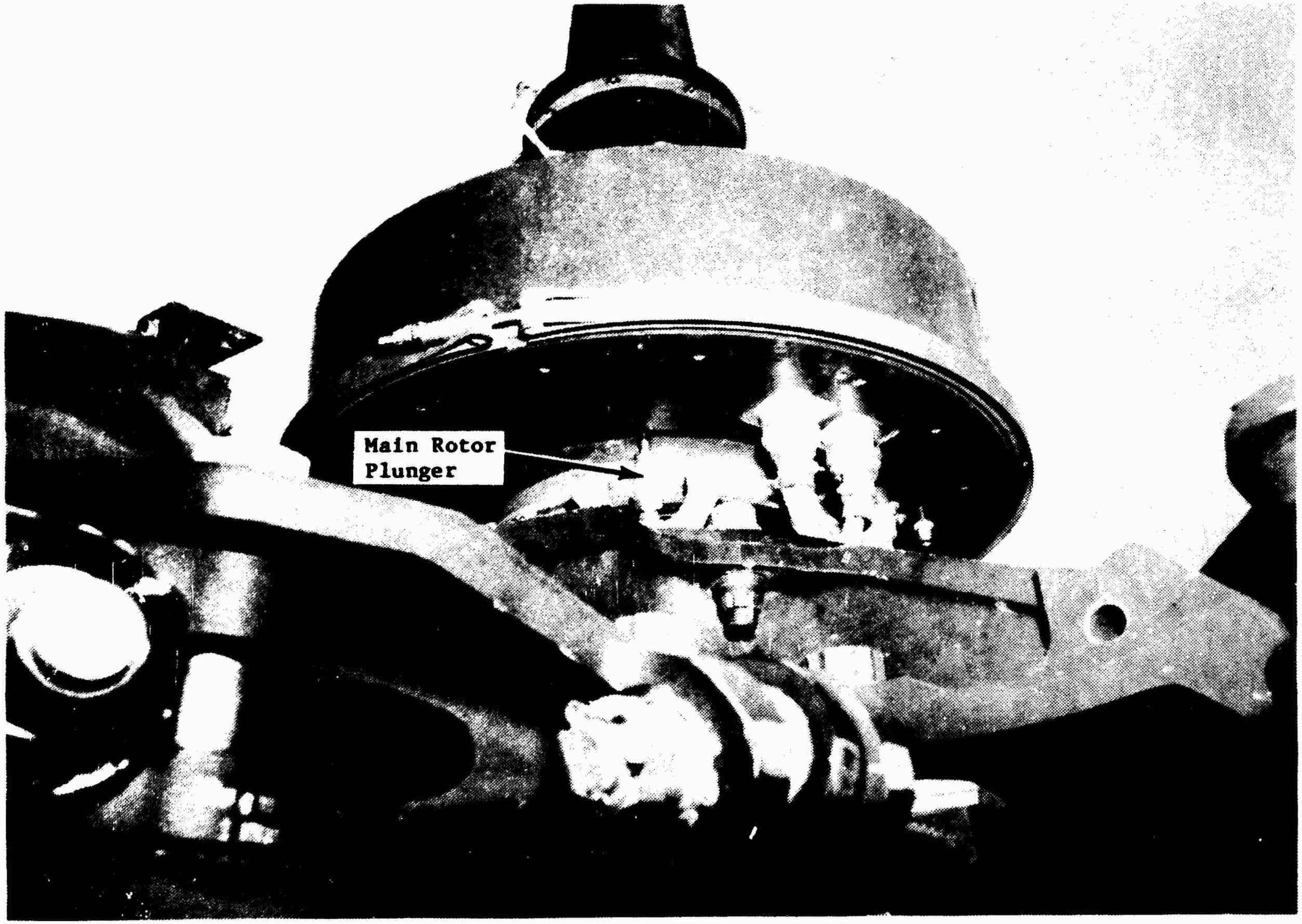


Photo 4. Main Rotor System

screws, nor does it check the position of the nut or retaining ring. During pilot training in preparation for this test, a portion of the main rotor plunger on the test aircraft broke off inside the main rotor assembly. Extensive maintenance was required to remove the broken piece and insure no internal damage had occurred to the main rotor assembly. Depression of the main rotor plunger on other production aircraft showed varying degrees of movement and failed to provide any conclusive information as to the position of the main rotor shaft nut and retention ring. The poor design of the main rotor bearing plunger assembly is a shortcoming. Consideration should be given to removing the main rotor bearing plunger assembly from production aircraft. The position of the main rotor shaft nut and retaining ring as well as torque on retaining screws should be determined by maintenance inspection procedures at the time of main rotor assembly and installation.

Engine Oil Level Access Doors

52. To open the engine oil level access doors required a common screwdriver. Without a tool to open the doors (photos 5 and 6), the oil level cannot be checked during the preflight inspection. Additionally, the doors are hinged so they open into the airflow, with the potential of being torn from the aircraft in flight if left unsecured. The poor design of the engine oil level access doors is a shortcoming.

Gearbox Grease Expulsion

53. During postflight inspections, large amounts of grease (approximately 6 ounces) were found on the tail rotor intermediate gearbox. The grease was apparently being forced out of the top breather port of the gearbox (EPR 84-10-10, app G). Since there are no provisions for servicing the intermediate gearbox (ref TM 55-1520-238-23), loss of grease will require replacement of the gearbox. Because of the repeated expulsion of grease, the gearbox was replaced 125.5 hours after its installation in the test aircraft. The replacement gearbox was modified with a baffle to prevent further loss of grease through the breather port. By the conclusion of this evaluation the new gearbox had accumulated 9.8 hours with no loss of grease. The excessive expulsion of grease from the top breather port of the intermediate gearbox is a shortcoming. The modified gearbox should be observed during further testing to evaluate the long-term effect of the modification.

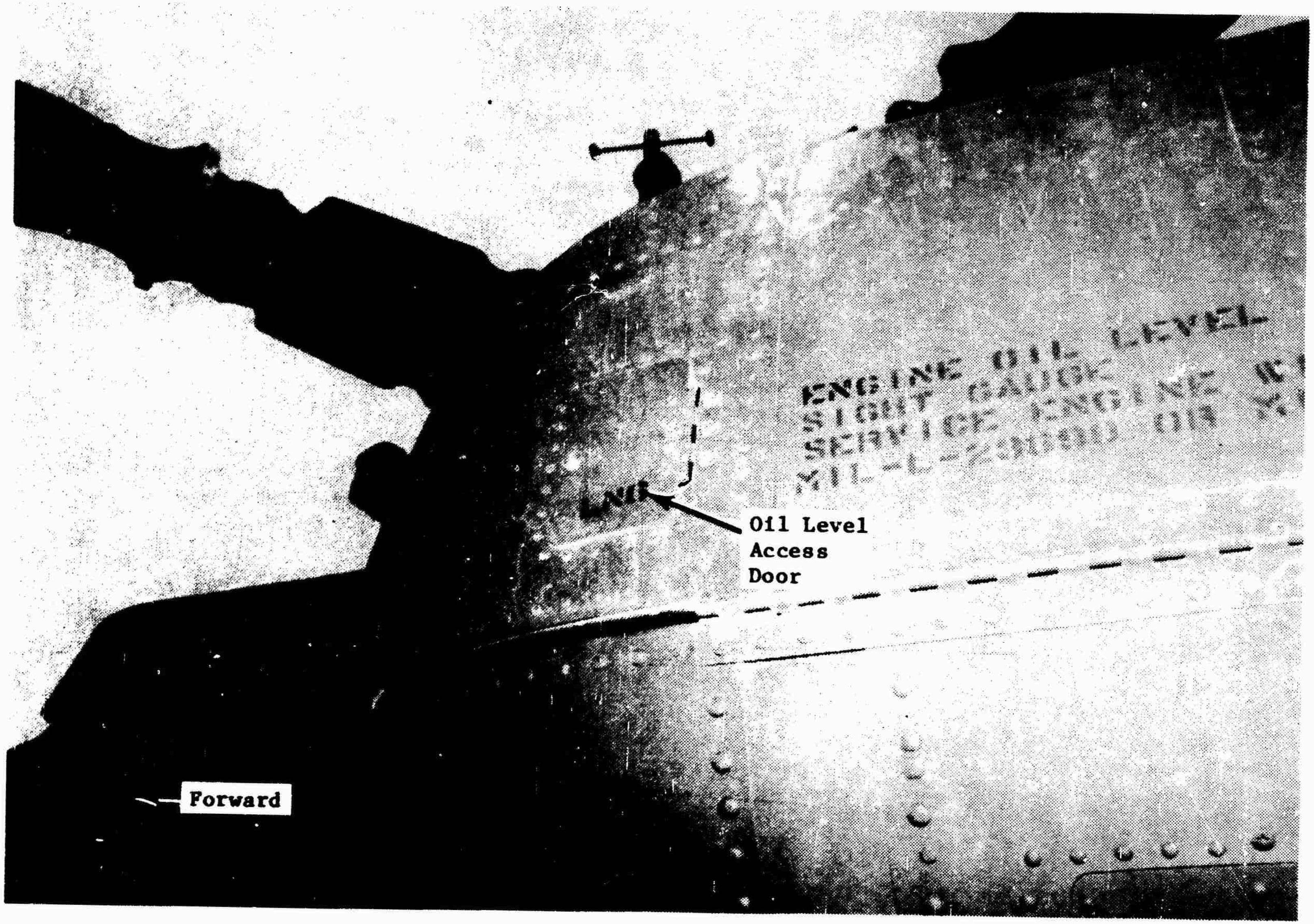


Photo 5. #1 Engine Oil Level Access Door

Photo 6. #1 Engine Oil Level Access Door

Electromagnetic Interference (EMI)

54. Throughout the evaluation, Electromagnetic Interference (EMI) in the caution panel was observed during ground and flight operations. During ground runup with the number one generator on line, moving the number two generator to the test position illuminated the "RECT" and "HOT BAT" lights and articulated the wing pylons. Moving the number one generator to test position resulted in the number one "RECT" and "MAN STAB" lights illuminating. Additionally, moving the engine instrument test switch to the test position would result in illumination of the number one "ENG OIL PSI" light. In flight, the master caution light would occasionally illuminate for one to two seconds. Due to the short illumination time, it was not possible to identify the associated caution panel light. The momentary master caution and caution panel light illumination was not repeatable or predictable. EMI in the caution panel is a shortcoming.

MISCELLANEOUS

55. Testing was accomplished to determine the status of shortcomings previously reported during A&FC 3. As a result of this evaluation, four were found to have been corrected. The following shortcomings were found to still exist:

a. The objectionable 4/rev vertical vibrations.

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

c. The poor anthropometric design of the pilot cyclic grip.

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

e. The poor design of the pilot fuel control panel.

f. The excessively large longitudinal cyclic inputs required when retrimming during IMC flight with attitude hold ON.

g. The lack of reliable indication of parking brake status.

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

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

j. The poor design of collective pitch control friction mechanism.

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

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

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

AIRSPEED CALIBRATION

56. The pilot and copilot pitot-static systems were calibrated in level flight and dives using a T-28 pace aircraft as a calibrated airspeed reference. The data are presented in figures 53 and 54, appendix E.

CONCLUSIONS

GENERAL

57. The following conclusions were reached upon completion of The First Article Preproduction Test of the AH-64A helicopter:

a. The level flight performance and handling qualities were not significantly different from the prototype.

b. The level flight performance presented in the operator's manual satisfactorily represents the performance of the production AH-64A (para 8).

c. One enhancing characteristic was found.

d. Two deficiencies were found, one of which was previously reported on the prototype YAH-64.

e. Four previously reported shortcomings have been corrected.

f. Twenty-six shortcomings were found, thirteen of which were previously reported on the prototype YAH-64.

g. Fifteen items of specification noncompliance were found.

h. Seventeen equipment performance reports were submitted.

ENHANCING CHARACTERISTIC

58. The automatic contingency power feature of the T700-GE-701 engine is an enhancing characteristic (para 22).

DEFICIENCIES

59. The following are deficiencies of the AH-64A (see app D for definition of deficiency used in this report):

a. False indication of engine failure (para 36).

b. The undesirable engine/airframe response during power applications from zero torque condition and during pull ups from less than 40% torque at less than one g (para 23).

SHORTCOMINGS

60. The following shortcomings (listed in order of relative importance) have been identified. Those shown with asterisks were reported during previous USAAEFA tests of the prototype YAH-64 and still exist (see app D for definition of shortcoming used in this report).

a. The objectionable 4/rev (19.3 Hz) vertical vibration characteristics (para 38).*

b. The restriction to the pilot's field-of-view caused by window edge distortion, the overhead circuit breaker panel, canopy reflection, CPG helmet, and the PNVS turret (para 55).*

c. The obstructed access to the pilot's #1 fire handle (para 40).

d. The failure of the HARS to consistently align with the correct magnetic heading (para 50).

e. The activation of the low rotor speed warning system at an excessively high rotor speed (96 to 97%) (para 37).

f. The lack of fuel transfer/crossfeed indications (para 41).

g. The poor anthropometric design of the pilot cyclic grip (para 42).*

h. The excessive accumulation of oil on the main transmission deck and in the upper fairing access area (para 55).*

i. The excessive expulsion of grease from the top breather port of the intermediate gearbox (para 53).

j. The poor design of the pilot fuel control panel (para 43).*

k. The excessively large longitudinal cyclic inputs required when retrimming during IMC flight with attitude hold ON (para 27).*

l. The poor design of the main rotor bearing plunger assembly (para 51).

m. The lack of reliable indication of parking brake status (para 55).*

n. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls (para 55).*

o. The lack of engine anti-ice/start bleed valve position indications (para 44).

p. The poor location of the pilot engine control quadrant (para 55).*

q. The poor design of the collective pitch control friction mechanism (para 14).*

r. The poor location of the tail wheel unlock light (para 55).*

s. The high inherent friction of the engine power control levers (para 15).*

t. The lack of reliable audio signal and azimuth information from the AN/ARN 89B ADF set (para 25).*

u. The lack of VDU roll attitude symbology to distinguish upward from downward direction (para 45).

v. The lack of individual light intensity adjustment of the CPG Marconi indicators (para 48).

w. The poor readability of the transponder code numbers (para 47).

x. The poor design of the engine oil level access doors (para 52).

y. The washout of the rocket panel display, Marconi instrument indications and caution, warning, and advisory panel segment lights in direct sunlight (para 55).*

z. The EMI in the caution panel (para 54).

SPECIFICATION NONCOMPLIANCE

61. The AH-64A was found to be not in compliance with the following paragraphs of the system specification (ref 1, app A). Additional specification noncompliance, beyond the scope of this evaluation may exist.

a. 10.3.2.1.2 - the longitudinal breakout force (plus friction) in the fore and aft direction with the CPG cyclic stick retracted, exceeded the maximum 10% asymmetry by 3.0% (para 11).

b. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the aft direction with the CPG cyclic stick retracted exceeded the 2.4 lb limit by 1.0 lb (para 11).

c. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the forward and aft directions with the CPG cyclic stick extended exceeded the 2.4 lb limit by 0.6 lb (para 11).

d. 10.3.2.5 - the longitudinal control force could not be trimmed to zero in flight with the CPG cyclic stick retracted (paras 11 and 28).

e. 10.3.2.1.3 - the lateral control breakout force (plus friction) exceeded the 1.5 lb limit by 1.0 lb left and 0.5 lb right (para 12).

f. 10.3.2.1.3 - the left directional breakout force (plus friction) exceeded the 7.0 lb limit by 2.0 lb (para 13).

g. 10.3.2.2 - the left control force versus displacement gradient was less than the minimum value of 2.0 lb/in. by 0.3 lb/in. (para 13).

h. 10.3.2.1 - the directional force gradient was not linear throughout the range of control travel and was concave upward instead of downward during the last 3/4 inch of control travel in each direction (para 13).

i. 10.3.2.1.3 - the collective breakout force (plus friction) in the up direction was lower than the minimum value of 3.0 lb by 0.4 lb (para 14).

j. 10.3.3.2.3 - the engine power control lever breakout force (plus friction) exceeded the maximum value by 2.5 lb (friction OFF) and 10.5 lb (friction ON) (para 15).

k. 10.3.4.1 - the static longitudinal stability was neutral about trim airspeeds of 70 KCAS in level flight and 81 KCAS in IRP climbs (para 17).

l. 10.3.8.1.1 - the collective control delay time following a single engine failure at maximum level flight airspeed was less than the minimum of 2.0 seconds by 1.0 second (para 35).

m. 3.2.11.5.1 - the 4/rev and 8/rev vertical vibration at the pilot station exceeded the specification maximum in certain airspeed ranges (para 38).

n. 3.7.7.1.3.10(c) - there is no advisory light to indicate when fuel transfer is in progress (para 41).

o. 3.7.5.4.1 - the failure of anti-ice/start bleed valve to close does not illuminate a caution light (para 44).

RECOMMENDATIONS

61. The following recommendations are made:

a. Correct the deficiencies prior to operational use (para 58).

b. Correct the shortcomings as soon as practical (para 59).

c. The following CAUTION should be placed in the operator's manual:

CAUTION

Rapid left roll inputs greater than 2 inches could exceed dual engine transient torque limits when operating above a trimmed torque setting of 85%.

d. The following statement should be added to the operator's manual: Intentional flight in IMC with attitude hold OFF is not recommended (para 28).

e. The operator's manual should be changed to prohibit intentional flight in IMC with DASE OFF (para 29).

f. Further testing of a fully equipped mission aircraft should be accomplished to insure adequate cooling air is available for the crew stations and the forward avionics bays simultaneously (para 46).

g. Further observation of the intermediate gearbox for grease expulsion should be accomplished (para 53).

APPENDIX A. REFERENCES

1. Systems Specification, *Hughes Helicopters*, DRC-S-H10000B, 15 April 1982, with all current specification change notices.

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

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

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

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

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

7. Final Report, USAAEFA Project No. 80-17-1, *Airworthiness and Flight Characteristics Part 1 YAH-64 Attack Helicopter*, September 1981.

8. Final Report, USAAEFA Project No. 80-17-2, *Aiworthiness and Flight Characteristaics Part 2 Advanced Attack Helicopter*, February 1982.

9. Final Report, USAAEFA Project No. 80-17-3, *Airworthiness and Flight Characteristics Test of AAH, Prototype Qualification Test-Government (PQT-G), Part 3 and Production Validation Test-Government (PVT-G) for Handbook Verification*, October 1982.

10. Letter, AVSCOM, DRSAV-ED, 23 March 1984, subject: Preliminary Airworthiness Evaluation of First Article Preproduction AH-64A Helicopter (Test Request).

11. Technical Manual, TM55-1520-238-10, *Operator's Manual for Army AH-64 Helicopter*, 28 June 1984.

12. Letter, AVSCOM, AMSAV-E, 23 July 1984, subject: Airworthiness Release for First Article Production Test - "Airworthiness Evaluation of the AH-64A Helicopter", USA S/N 82-23355.

13. Letter, Hughes Helicopter Inc., USAARPRO, 30 July 1984, subject: Operating Procedures Due to Deviations/Changes.

14. AMC Pamphlet, AMCP 706-204, Engineering Design Handbook, *Helicopter Performance Testing,* 1 August 1974.

15. Flight Test Manual, Naval Air Test Center, FTM 101, *Helicopter Stability and Control,* 10 June 1968.

APPENDIX B. DESCRIPTION

TABLE OF CONTENTS

GENERAL

1. The AH-64 Advanced Attack Helicopter (fig. 1) is a tandem, two-place, twin turbine engine, single main rotor aircraft manufactured by Hughes Helicopter Incorporated (HHI). The main rotor is a four-bladed fully articulated system supported by a stationary mast which transmits flight loads directly to the fuselage. The tail rotor is a dual-teetering, semi-rigid, delta-hinged system incorporating elastomeric flapping bearings. The rotors are driven by two General Electric T700-GE-701 engines through the power train shown in figure 2. An AiResearch GTCP 35-55 (H) auxiliary power unit (APU), is installed to drive the accessory section of the main transmission when the rotors are not turning. This provides pneumatic power for engine starting as well as electrical and hydraulic power for aircraft systems. The aircraft is designed to carry ordnance stores internally in the ammunition bay and externally on four wing store positions. The AH-64 is designed to operate during day, night, and marginal weather combat conditions using the Martin-Marietta Target Acquisition Designation System (TADS)/Pilot Night Vision Sensor (PNVS). The test aircraft, SN 82-23355, was configured with an aerodynamic mockup of the TADS/PNVS, 30mm chain gun, and eight Hellfire missles to represent the primary mission configuration (photos 1 through 8). The major modifications and external configuration differences between the YAH-64 (AV05) flown during the airworthiness and flight characteristics (A&FC) Part 3 tests and the aircraft configuration are presented in table 1. Drag estimates provided by HHI are included in this table for the items not on the test aircraft but described in the system specification for the program and for any external test instrumentation equipment. The backup control system (BUCS) was not operational for this test.

Table 1. Configuration of Test Aircraft (USA S/N 82-23355)

Configuration differences from prototype aircraft (AV05) flown during A&FC Part 3:

a. Installed digital automatic stabilization equipment (DASE) production software (program no. 7-211 D00005-11)

b. Wing pylons shortened 6 inches

c. Maintenance step installed on left side of aircraft

Items not on the test aircraft but defined in the system specification for the production program:

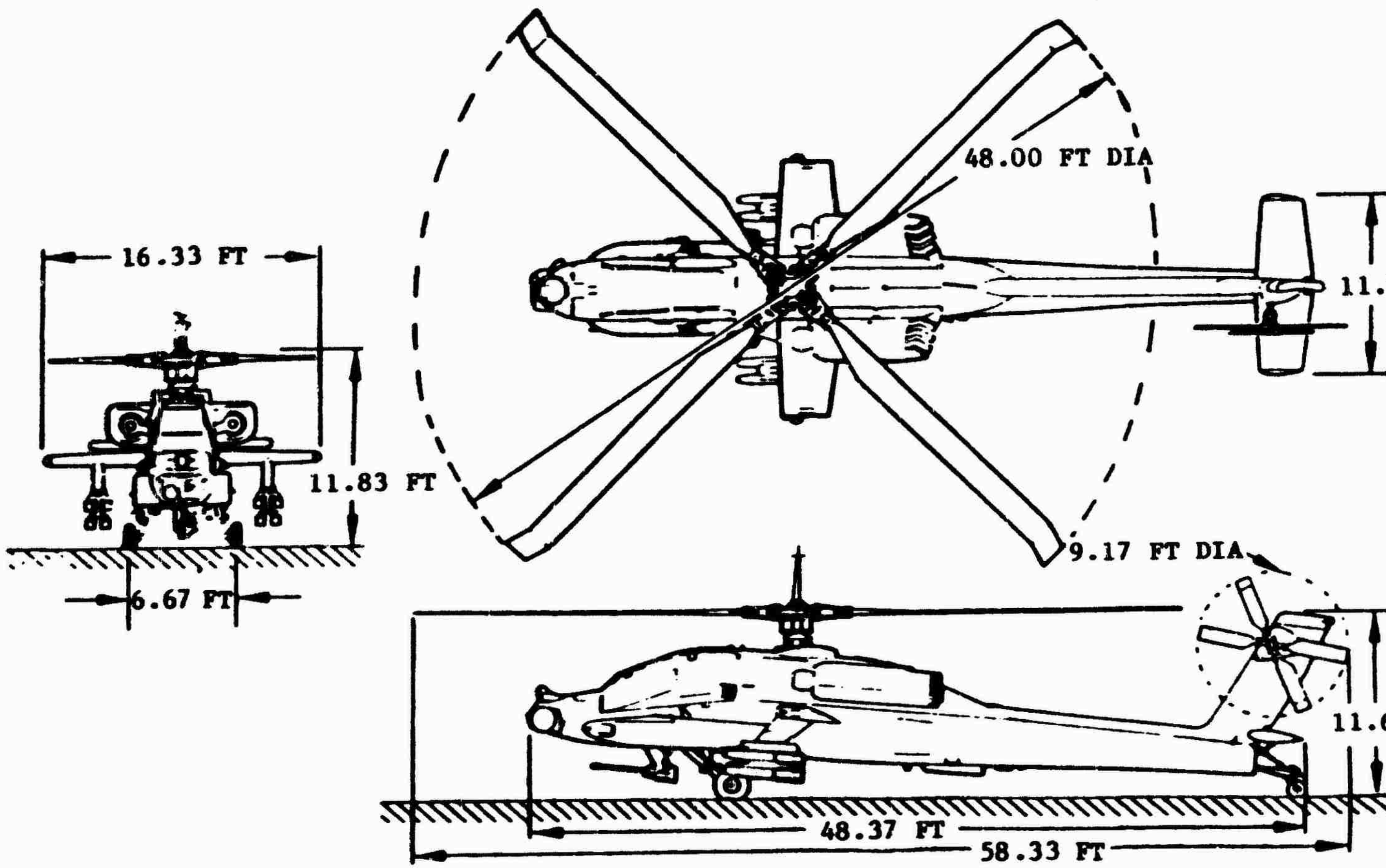


Figure 1. Aircraft Dimensions

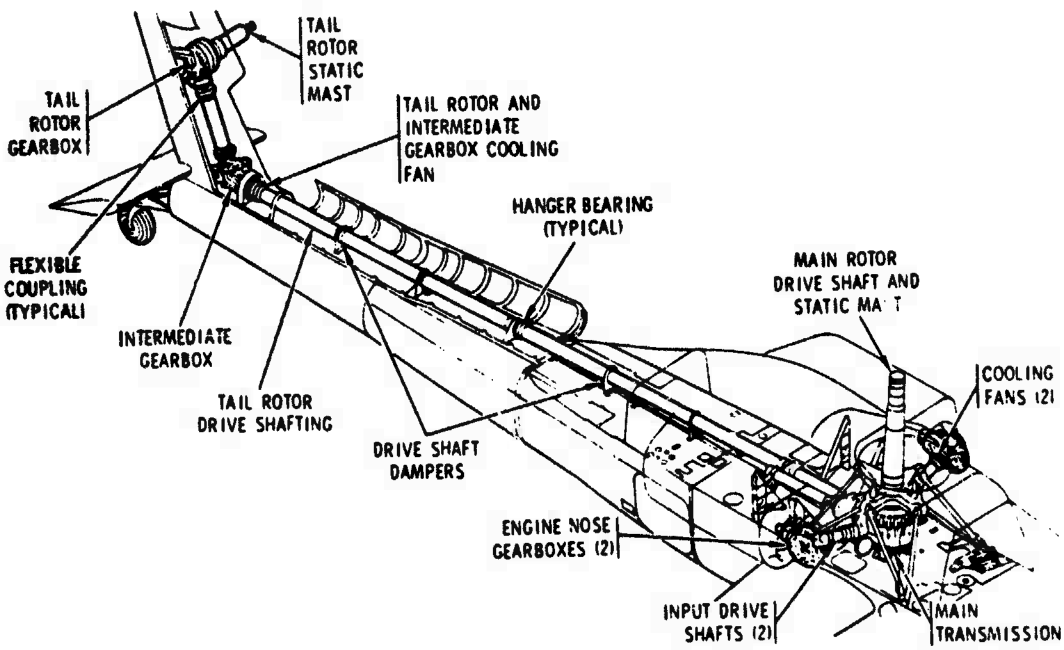


Figure 2. Powertrain

Photo 1. Front View, AH-64A (Apache) Helicopter

Photo 2. Right Quarter View, AH-64A (Apache) Helicopter

Photo 3. Right View, AH-64A (Apache) Helicopter

Photo 4. Right Three-Quarter View, AH-64A (Apache) Helicopter

Photo 5. Rear View, AH-64A (Apache) Helicopter

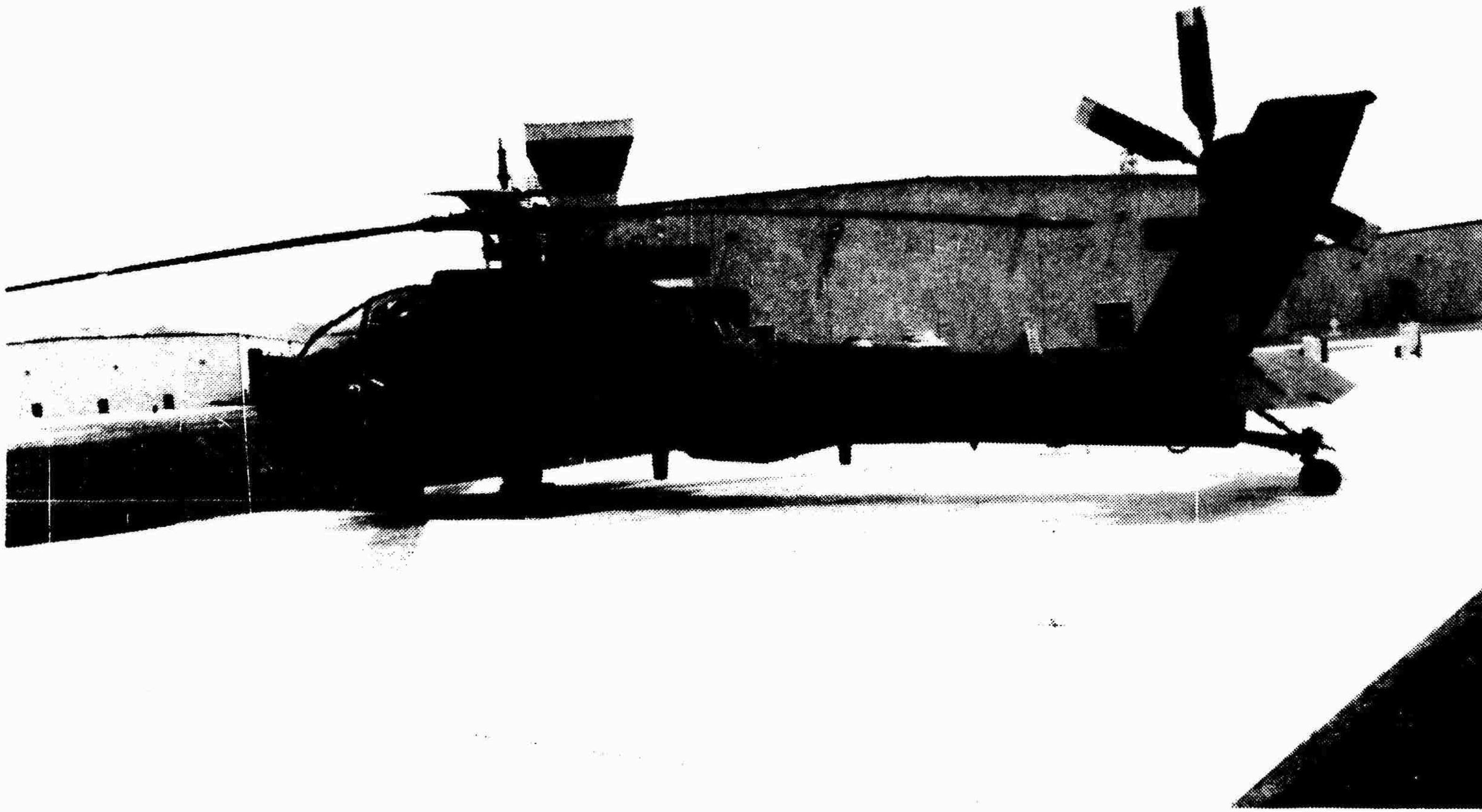


Photo 6. Left Three-Quarter View, AH-64A (Apache) Helicopter

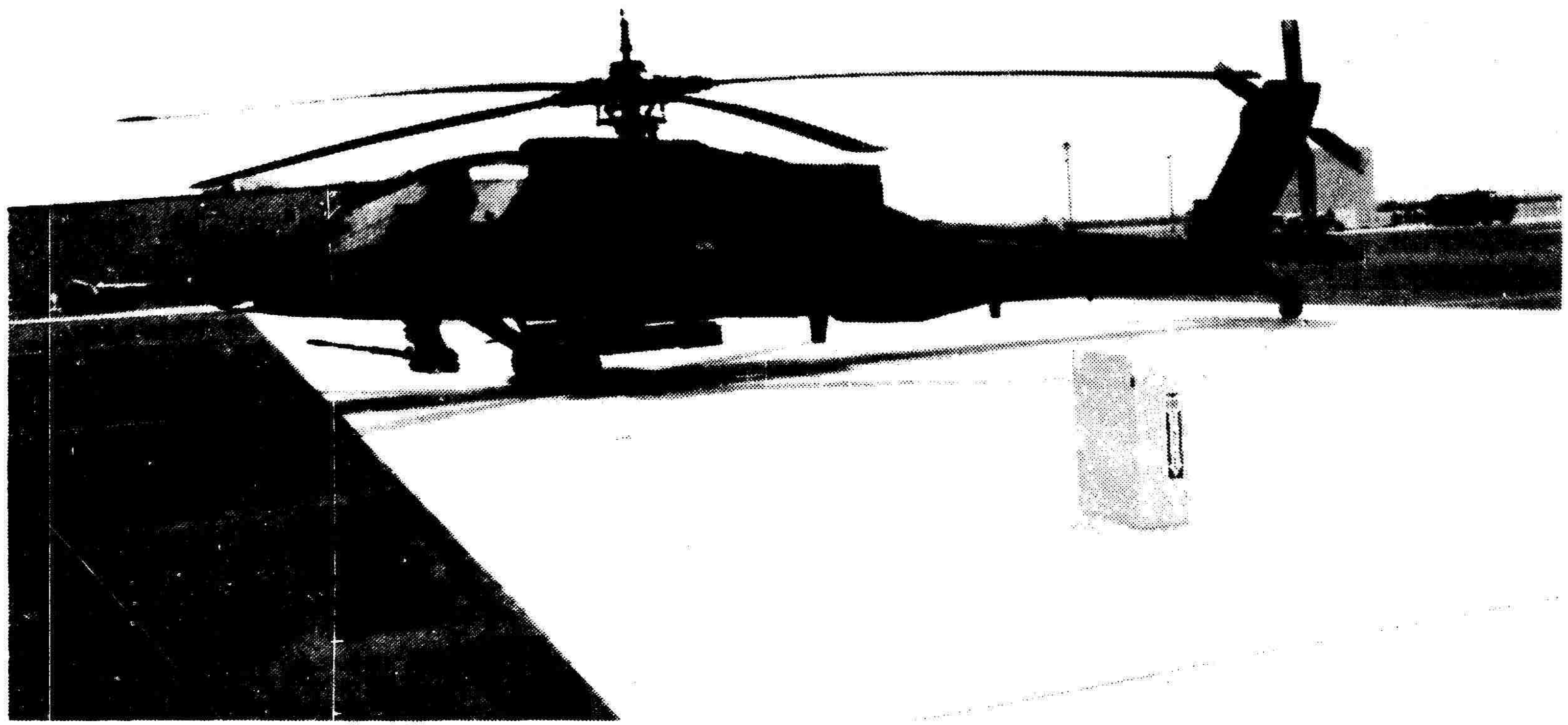


Photo 7. Left View, AH-64A (Apache) Helicopter

Photo 8. Left Quarter View, AH-64A (Apache) Helicopter

a. 30mm ammunition chute

b. Fire control computer

Items not in the system specification but installed on the test aircraft:

	Estimated Drag ΔFe (SQ FT)
a. Airspeed boom with angle of attack and sideslip vanes	0.90
b. Instrumentation canister located on main rotor hub	0.14
c. Telemetry antenna brackets located on scissors assembly and telemetry antenna and bracket located on underside of tail cone.	0.20
d. Instrumented main and tail rotor pitch change links	0.01
e. Instrumented main rotor blade	
f. Instrumented tail rotor blade	

DIMENSIONS AND GENERAL DATA

Main Rotor

Diameter (ft)	48.0
Blade chord (in.)	21.0*
Main rotor total blade area (sq ft)	166.5
Main rotor disc area (sq ft)	1809.56
Main rotor solidity (thrust weighted no tip loss)	0.092
Airfoil	HH-02**
Twist (deg)	-9
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
Gear ratio (engine to main rotor)	72.424322

*Includes tips
**Outer 20 inches swept 20 degrees and transitioned to a NACA 64A006 airfoil

Tail Rotor

Diameter (ft)	9.17
Chord; constant (in.)	10
Tail rotor total blade area (sq ft)	14.89
Tail rotor disc area (sq ft)	66.0
Tail rotor solidity	0.2256
Airfoil	NACA 63-414 (modified)
Twist	-8.8
Number of blades	4
Rotor speed at 100 percent N_R (rpm)	1403.4
Distance from main rotor mast centerline (CL) (ft)	29.67
Tip speed at 100 percent N_R (ft/sec)	673
Teetering angle (deg)	10 soft/15 Hard
Maximum blade angle (deg)	27

Horizontal Stabilator

Weight (lb)	77.3
Area (ft)	33.36
Span (ft)	11.17
Tip chord (ft)	2.65
Root chord (ft)	3.67
Geometric aspect ratio	3.41
	Variable (35 degrees) leading edge up to 10 degrees leading edge down)
Sweep of leading edge (deg)	2.89
Sweep of trailing edge (deg)	-7.23
Dihedral (deg)	0

Vertical Stabilizer

Area (from boom CL) (sq ft)	32.2
Span (from boom CL) (in.)	113.0
Root chord (at boom CL) (in.)	44.0
Geometric aspect ratio	2.75
Airfoil	NACA 4415 (modified)
Sweep of leading edge (deg)	28.1
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
Airfoil at root	NACA 4423

Aircraft

Fuel quantity (gal)	375
Basic design gross weight (lb)	14,660
Maximum gross weight (lb)	17,650
Primary mission gross weight (lb)	14,694
Maximum alternate gross weight (lb)	17,650
Extended range gross weight	20,100

FLIGHT CONTROL DESCRIPTION

General

2. The AH-64A helicopter employs a hydromechanical irreversible flight control system which 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 stability augmentation system (SAS) actuators. Hydraulic power is supplied by two independent 3000 psi hydraulic systems which are powered by hydraulic pumps mounted on and driven by the accessory gearbox of the main transmission to allow full operation under a dual-engine failure condition. A DASE system is installed to provide rate damping and command augmentation. The DASE is limited to ± 10 percent of control authority in roll and yaw. The longitudinal cyclic hydraulic servoactuator allows 20 percent forward and 10 percent aft control authority in the pitch axis. The DASE also provides attitude hold and a Hover Augmentation System (HAS). An electrically actuated horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. Movement of the stabilator can be controlled either manually or automatically. A trim feel system is incorporated in the cyclic and directional controls to provide a control force gradient with control displacement from a selected trim position. A 3 position trim release switch located on the pilot's cyclic grip, provides discrete off, or momentary interruption of the force gradient in all axes simultaneously to allow the cyclic or pedal controls to be placed in a new reference trim position. Full control travel is 10.5 inches in the longitudinal control, 8.45 inches in the lateral control, 11.73 inches in the collective control and 6.06 inches in the directional pedals.

Cyclic Control System

3. The cyclic control system (fig. 3) consists of dual-tandem cyclic sticks attached to individual support assemblies in each cockpit. The support assembly houses the primary longitudinal and lateral control stops, and two linear variable differential transducers (LVDTs) designed to measure electrically the longitudinal and lateral cyclic position for the DASE computer inputs. A series of push-pull tubes and bellcranks transmit 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 main rotor pitch change links through the rotating swashplate (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 weapons systems.

Collective Control System

4. The collective pitch control system (fig. 6) consists of dual-tandem collective sticks which transmit collective 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 from the collective servoactuator to the load demand spindle of each engine hydromechanical unit (HMU). The HMU meters the fuel as appropriate to provide compensation for collective pitch. Located at each collective control base assembly are the primary control stop, an LVDT, and a one g balance spring. Redundant LVDTs on the collective servoactuator supply electrical inputs to the stabilator control units.

5. Each collective 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 both engines 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 transmit directional pedal inputs to the tail rotor hydraulic servoactuator located in the vertical stabilizer. Attached to each directional pedal assembly is a primary tail rotor control stop and one LVDT. Two sets of wheel brake cylinders are attached to the directional pedals and a 360 degree swiveling tail wheel is incorporated.

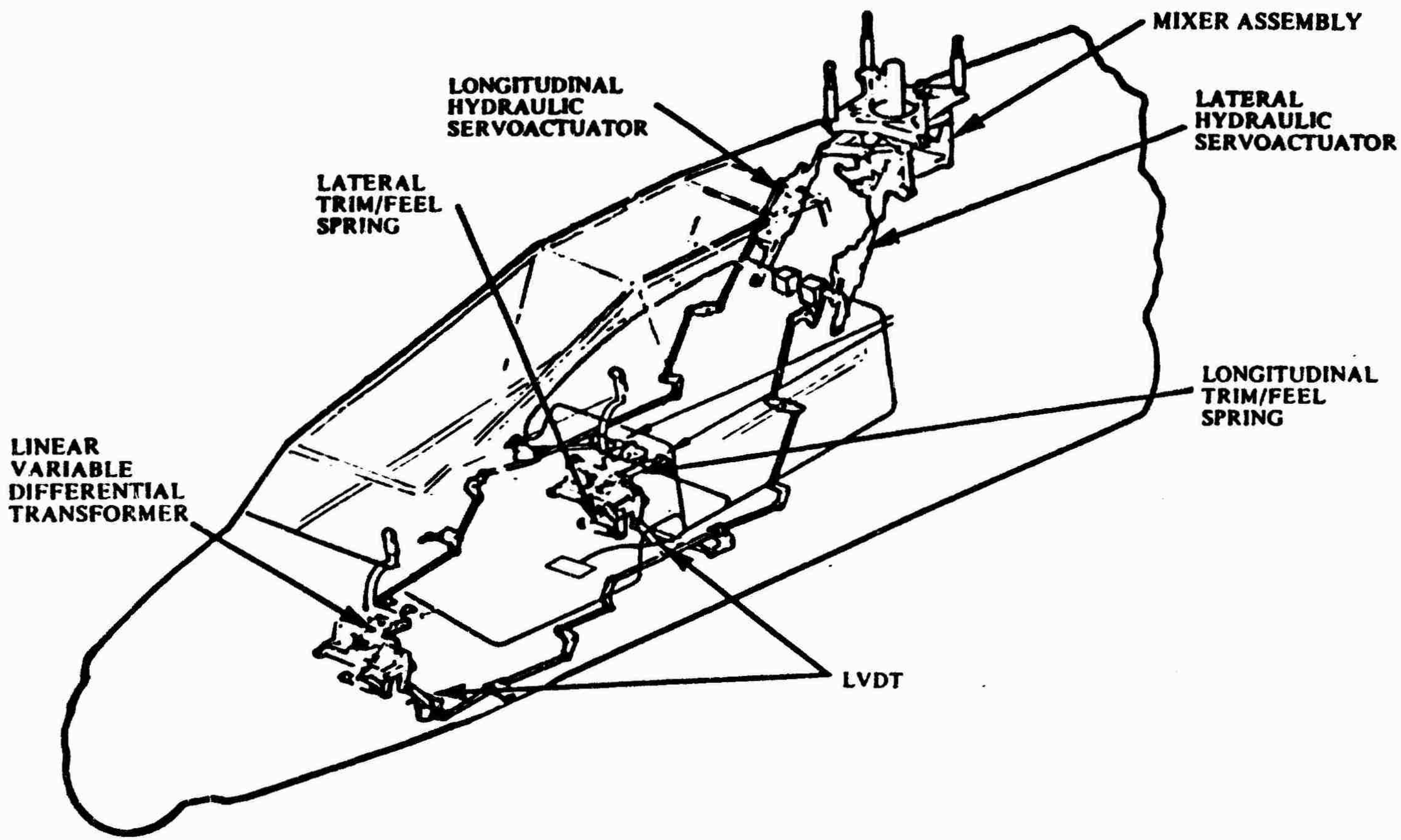


Figure 3. Cyclic Control System

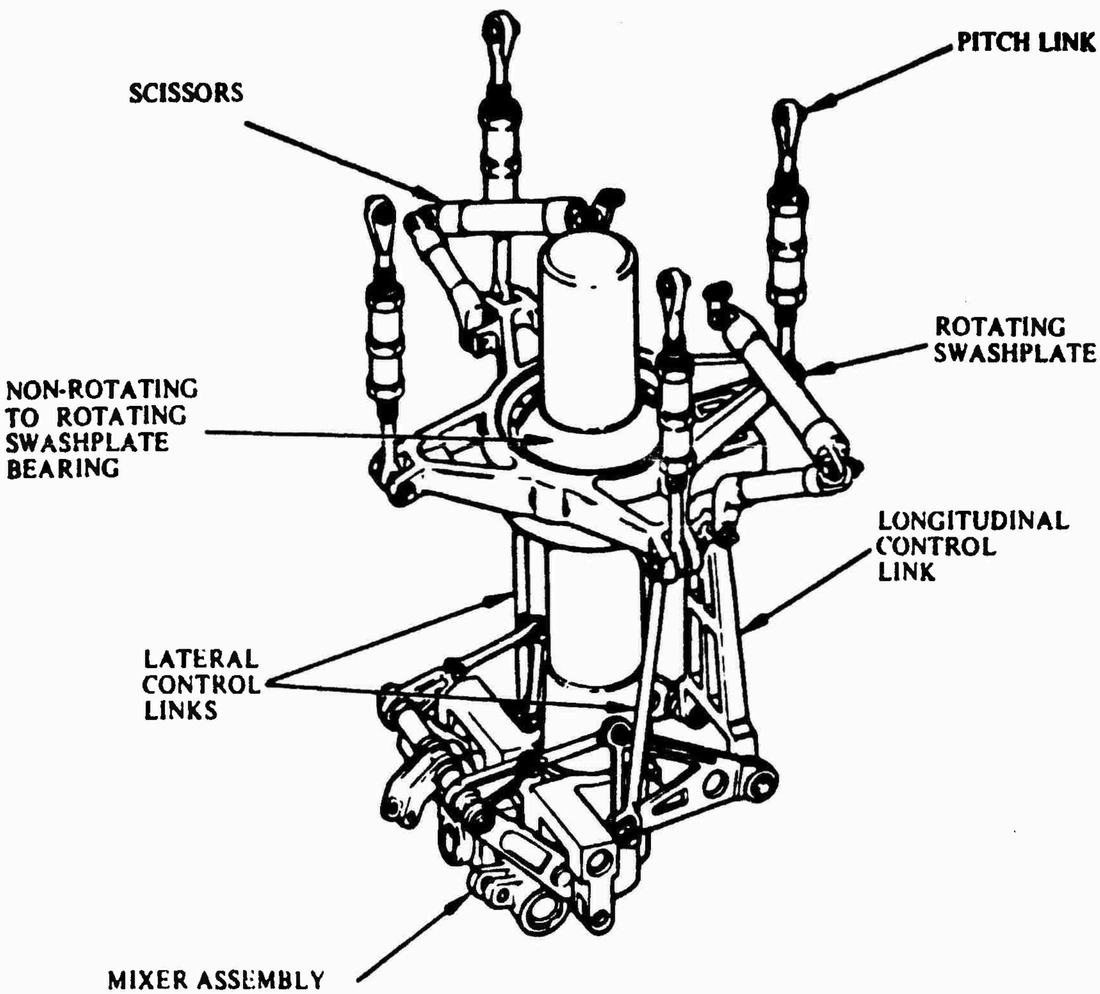
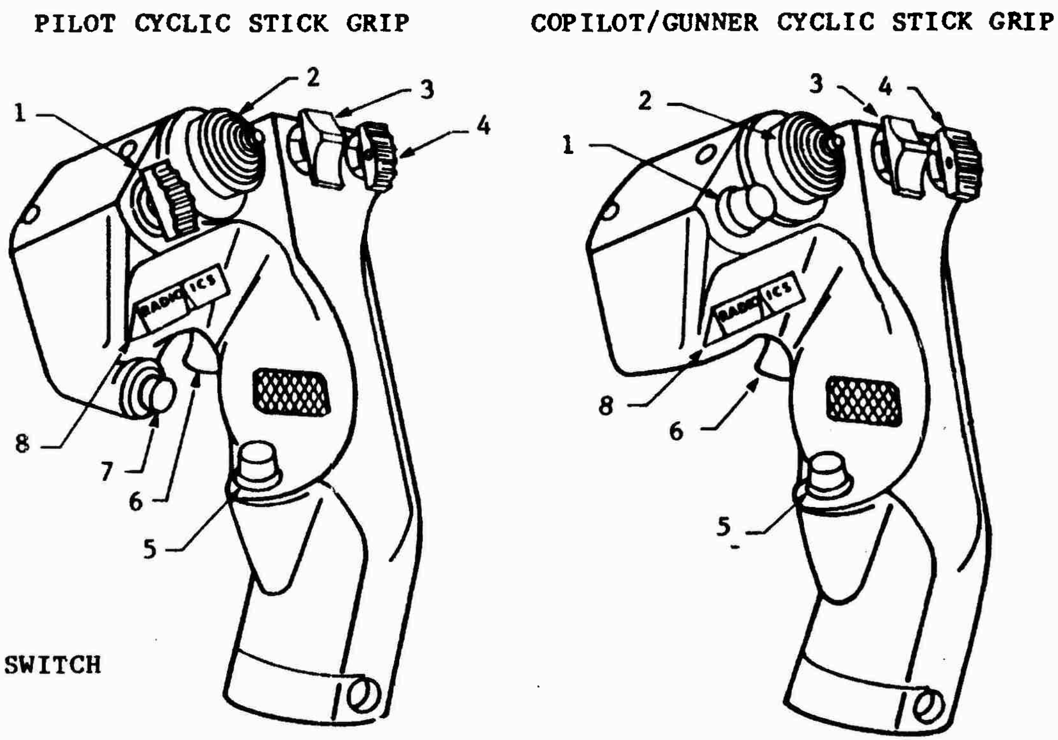


Figure 4. Main Rotor Swashplate Assembly



1 TRIM RELEASE SWITCH
(MOMENTARY ONLY CPG)

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

Figure 5. Cyclic Stick Grips

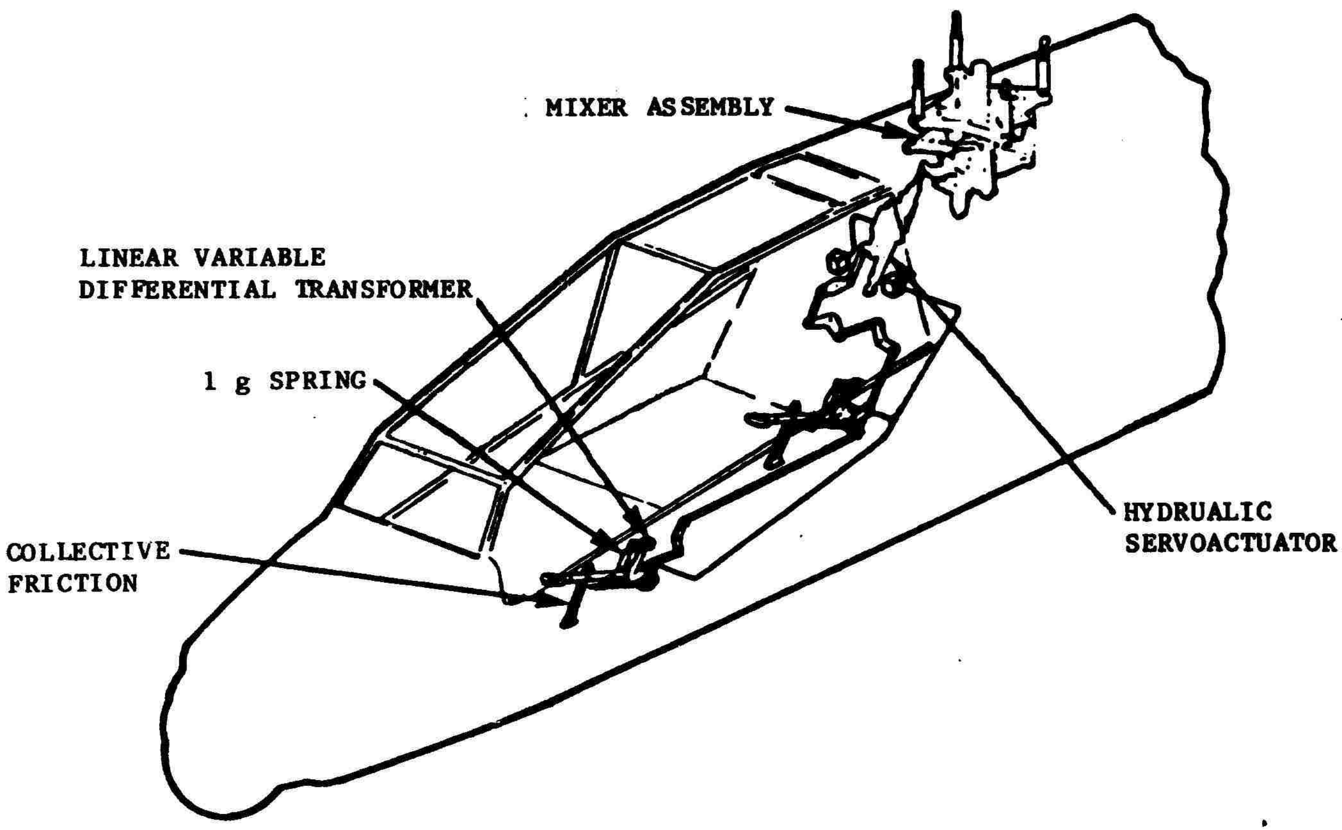
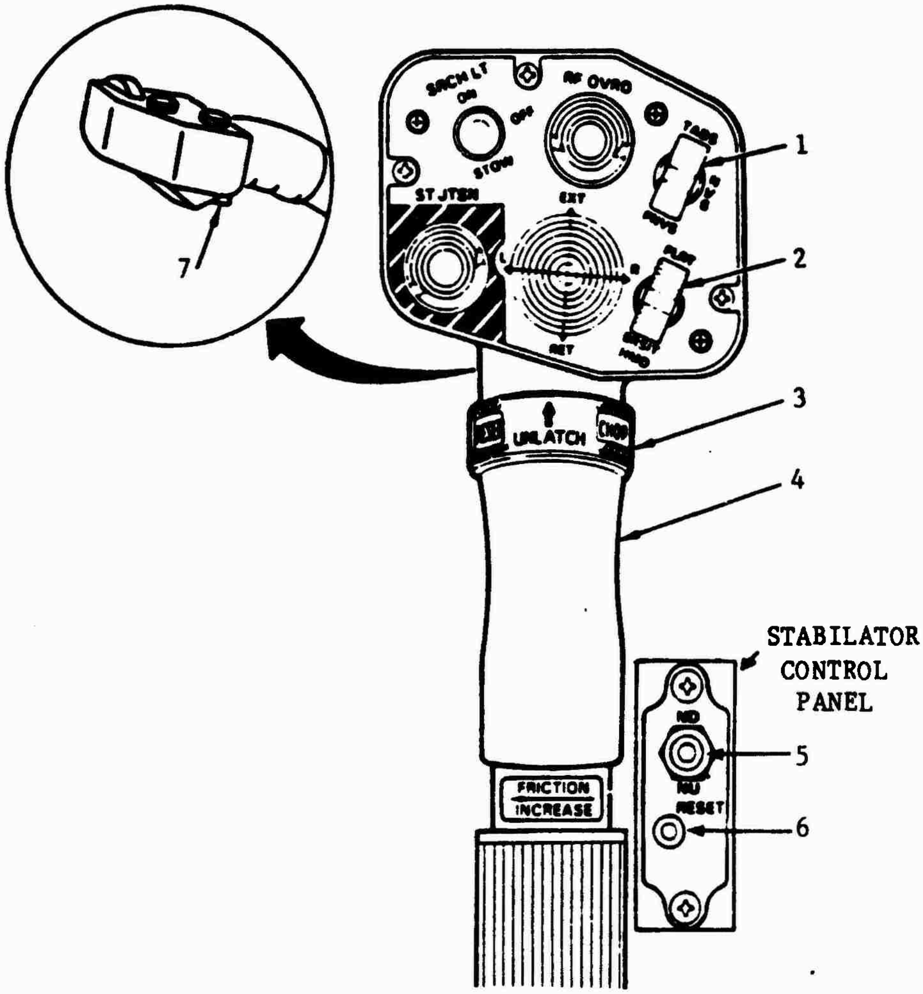


Figure 6. Collective Control Subsystem



PILOT AND CPG COLLECTIVE STICK

1 NIGHT VISION SWITCH

2 BORESIGHT HMD SWITCH

3 ENGINE CHOP COLLAR

4 COLLECTIVE GRIP

5 STABILATOR MANUAL CONTROL SWITCH

6 AUTOMATIC OPERATION/AUDIO WARNING
RESET BUTTON

7 BUCS TRIGGER (CPG ONLY)

Figure 7. Collective Stick Control

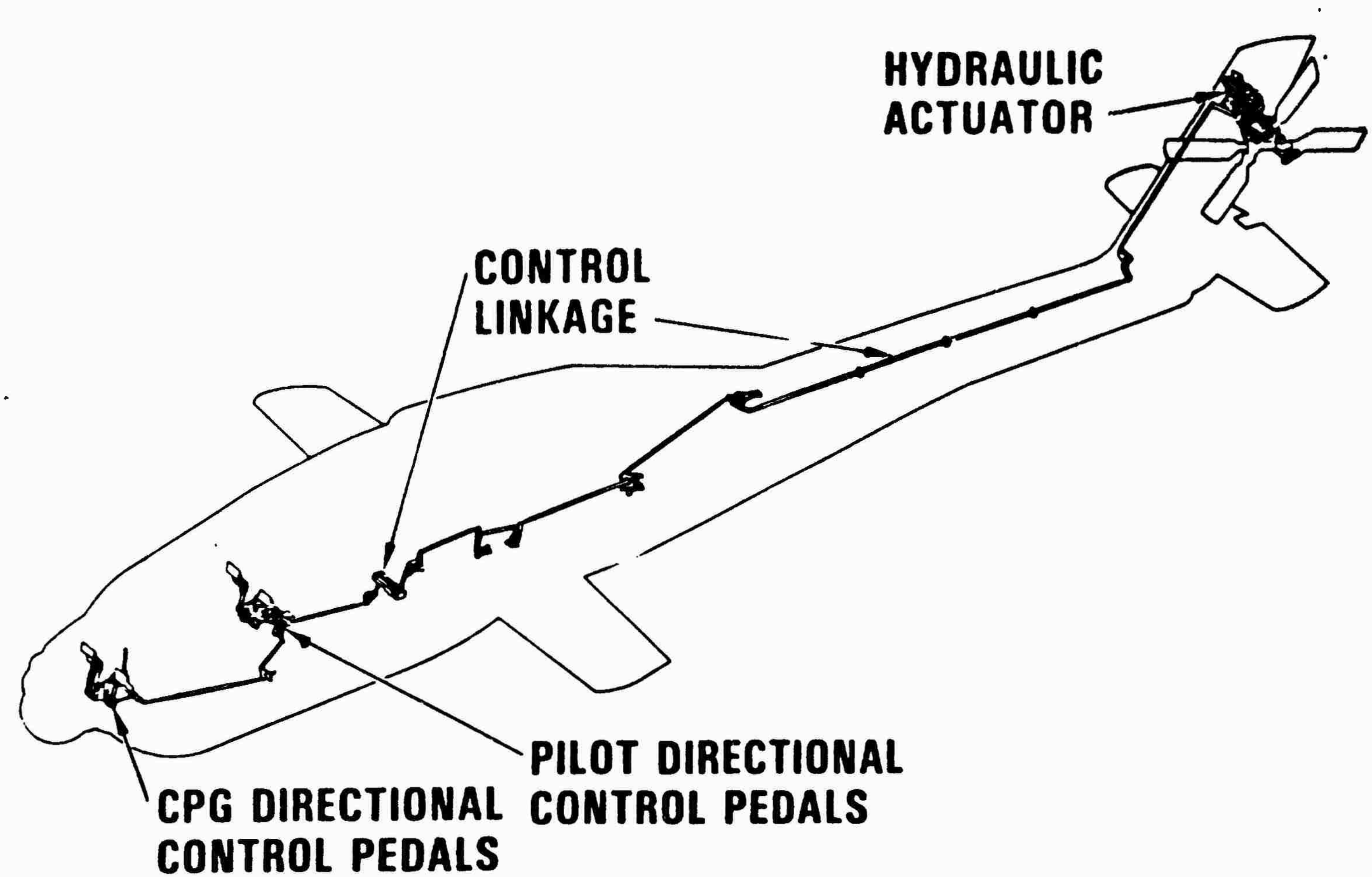


Figure 8. Directional Controls

The tail wheel may be locked in the trailing position by means of a switch located on the pilot instrument panel.

Trim Feel System

7. A trim feel system is incorporated in the longitudinal, lateral, and directional control systems. The system 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 trim feel system and allow the cyclic and directional pedals to move freely without resistance from the trim feel springs. The trim release switch on the CPG grip allows momentary release of the trim feel system. A three position trim release switch on the pilot's grip allows the system to be ON, discrete OFF, or momentarily released.

Horizontal Stabilator

8. A horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. A block diagram of the stabilator control system is shown in figure 9. A dual series, 28 VDC electromechanical actuator allows incidence changes of +35 trailing edge down (TED) to -10 degrees TED 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. There are three modes of stabilator operation: the automatic mode, the Nap-of-the-earth (NOE)/Approach mode and the manual mode. The stabilator is controlled in the automatic mode by two stabilator control units (SCUs). Each SCU controls one side of the dual actuator. Both SCUs receive collective control position information from redundant LVDTs located on the collective servoactuator. Two independent pitch rate gyros provide pitch rate information to the SCUs (one gyro for each SCU). The Air Data System (ADS) provides airspeed to both SCUs. Additionally, the left-hand pitot-static system supplies airspeed to one SCU and the right-hand system provides airspeed to the other SCU. Both SCUs receive position information from both sides of the dual actuator.

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. Software in the SCU limits the incidence change in the automatic mode from +25 deg to -5 deg TED.

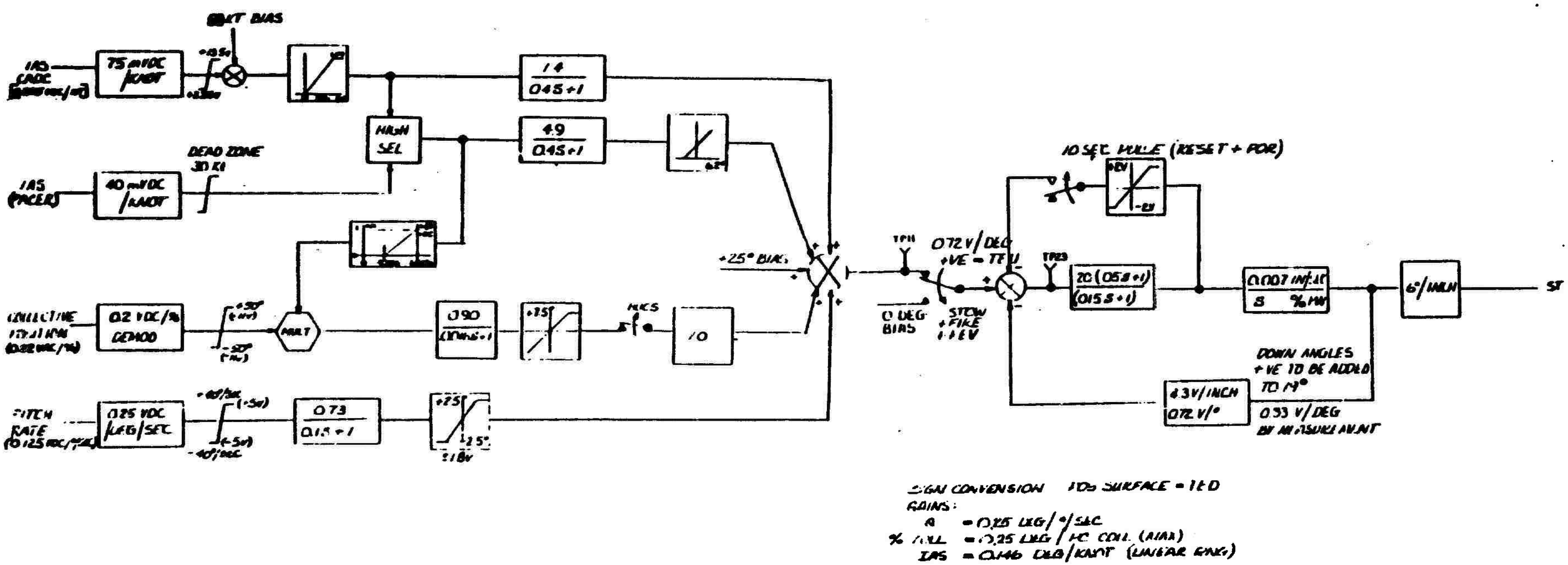


Figure 9. Stabilator Control Block Diagram

10. The NOE/Approach Mode is selected through the NOE/APPR mode switch on the pilot DASE panel. This mode is selectable at any airspeed when the automatic mode is engaged. The mode becomes operational below 80 knots indicated airspeed (KIAS) and will move the stabilator to 25 degrees TED. The NOE/approach mode can be disengaged by manual mode selection below 80 knots, activation of the DASE release, by the AUTO STAB reset switch or by turning the NOE approach mode switch off. Acceleration through 80 KIAS will engage the automatic schedule and the stabilator will move to the automatic mode schedule position. Failure to revert to automatic schedule will result in system disengagement with both visual and aural indications.

11. The manual mode can be selected below 80 KIAS through the pilot and CPG manual control switch on either collective stick. Manual control selection will result in MAIN STAB caution light illumination. Selection of automatic mode can be accomplished by pressing the AUTO STAB reset switch on the pilot or CPG collective stick. Acceleration through 80 KIAS in manual mode will engage the automatic mode and the stabilator to the automatic mode schedule position.

Flight Control Rigging

12. A flight control rigging end to end check was performed in accordance with procedures outlined in HHI Production Acceptance Test Procedure PATP 00007. The horizontal stabilator schedule is shown in figure 10. Tables 2 and 3 present the collective and cyclic rigging. Tail rotor rigging is shown below:

Full right pedal: 14.5 degrees thrust to left
Full left pedal: 27.9 degrees thrust to right

Digital Automatic Stabilization Equipment

13. The DASE provides rate damping (SAS), command augmentation (CAS), HAS, attitude hold, and turn coordination. A block diagram of the DASE is provided in figure 11. 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, roll, and heading), and inertial horizontal and lateral velocities (measured by the Doppler radar). The ADS provides longitudinal airspeed and sideslip angle. The LVDTs provide longitudinal, lateral, collective and directional control position information. The DASEC processes this information and commands control inputs

FIGURE 10
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

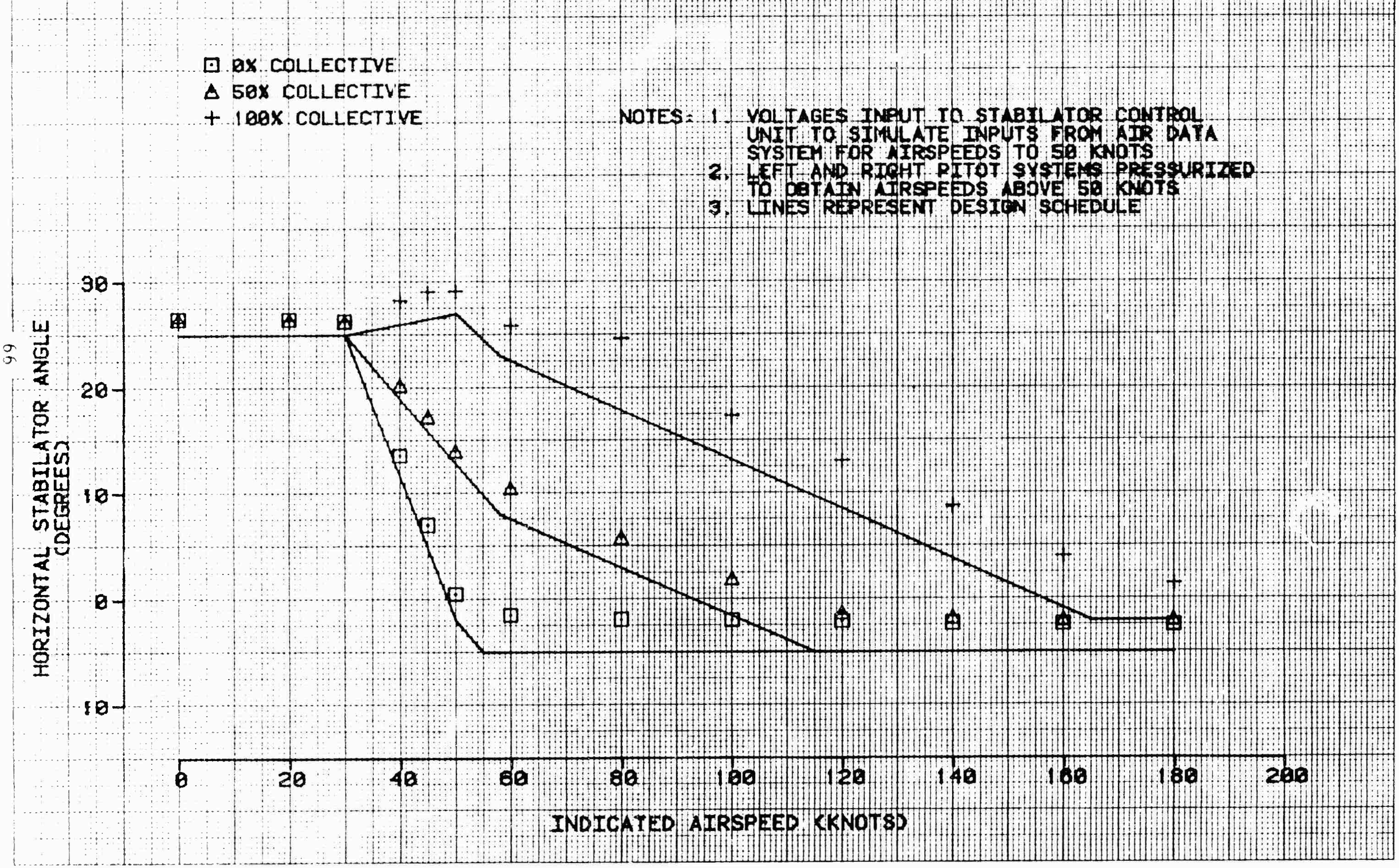


Table 2. Angle Measurements - Collective and Cyclic Controls

Blade Azimuth Position (deg)	Item	Rig Pins			Stick position			Pitch Housing Angle	Leading Edge Up or Down
		Collective	Longitudinal Cyclic	Lateral Cyclic	Collective	Longitudinal	Lateral		
$\psi = 90$	1	In	In	In	Rig	Rig	Rig	0.6	Up
	2	In	Out	In	Rig	Fwd	Rig	19.6	Down
	3	In	Out	In	Rig	Aft	Rig	11.35	Up
	4	In	In	In	Rig	Rig	Rig	0.45	Up
	5	Out	In	In	Up	Rig	Rig	10.7	Up
	6	Out	In	In	Down	Rig	Rig	8.0	Down
	7	In	In	In	Rig	Rig	Rig	0.58	Up
$\psi = 180$	8	In	In	In	Rig	Rig	Rig	0.1	Up
	9	In	In	Out	Rig	Rig	Left	11.0	Down
	10	In	In	Out	Rig	Rig	Right	7.9	Up
	11	In	In	In	Rig	Rig	Rig	0.1	Up

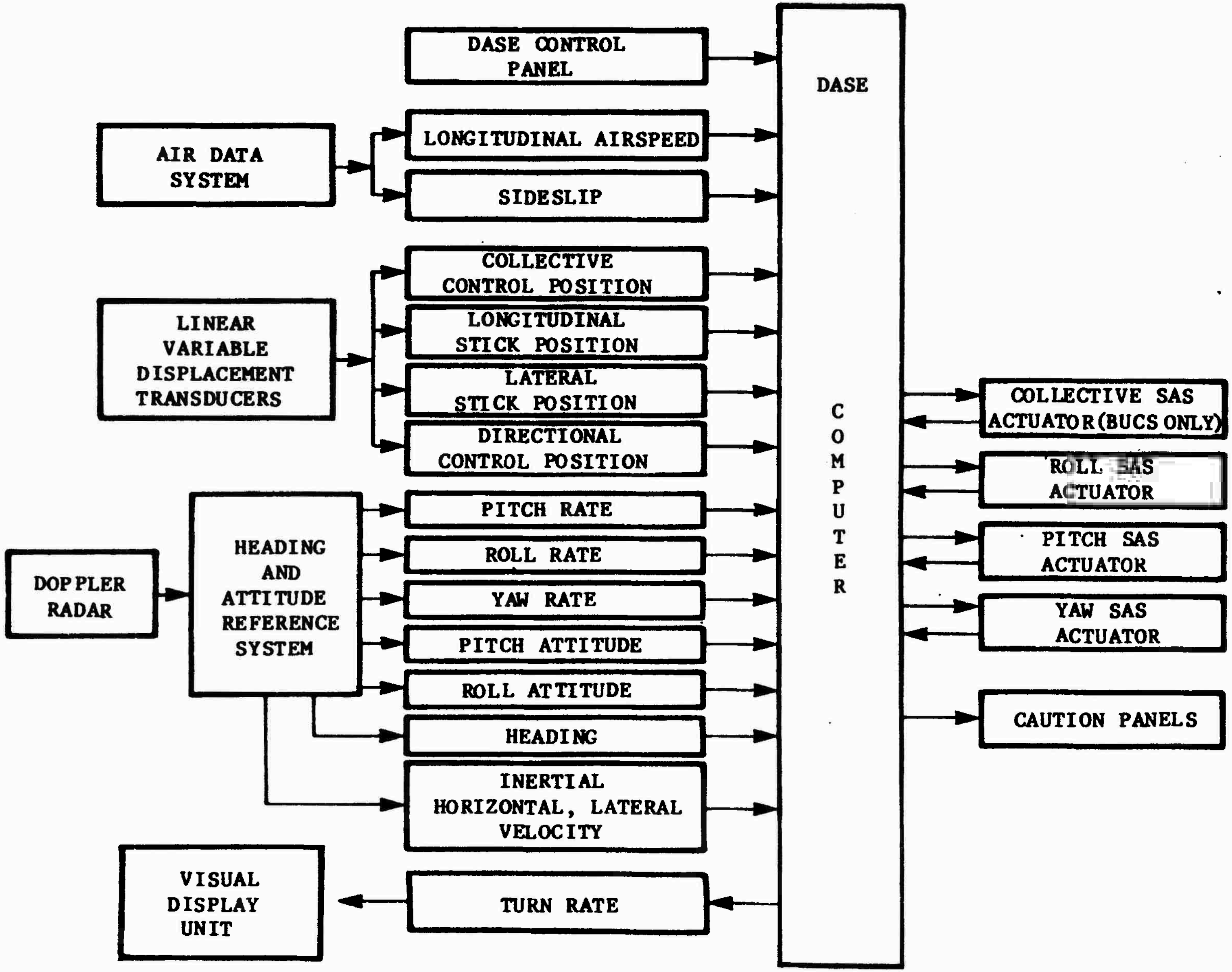


Figure 11. Digital Automatic Stabilization Equipment Block Diagram

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

Computation	Travel (deg)	Requirements (deg)
LONGITUDINAL CYCLIC		
1. Forward (Item 2 - Item 1)= (If Item 1 is leading edge up, add item 2 to Item 1)	20.2	20 (minimum)
2. Aft (Item 3 - Item 1) = (If Item 1 is leading edge down, add Item 3 to item 1)	10.8	10 (minimum)
LATERAL CYCLIC		
3. Left (Item 9 - Item 8) = (If Item 8 is leading edge up, add Item 9 to item 8)	11.1	10.3 (minimum)
4. Right (Item 10 - Item 8) = (If Item 8 is leading edge down, add Item 10 to item 8)	7.7	7 (minimum)
COLLECTIVE		
5. Full pitch travel (Item 5 - Item 6)= (If Item 6 is leading edge down, add Item 5 to item 6)	18.7	18 (minimum)
6. Collective pitch full down	-8	
Measured at pitch housing		-10 to -7

through the electrohydraulic servo valves on the longitudinal, lateral, and directional servoactuators. The DASE authority is limited in the lateral and directional axes to ± 10 percent of full control authority while the longitudinal axis is limited to 20 percent forward and 10 percent aft.

14. 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 control panel shown in figure 12. The CAS is used to augment the pilot control inputs and is an automatic function of the DASE whenever pitch and roll SAS are selected and yaw SAS is selected below 60 knots true airspeed (KTAS). The yaw CAS function is automatically disengaged during ground operations. Schematic diagrams showing gains and transfer functions of SAS/CAS are provided as figures 13 through 17.

15. A limited authority HAS mode is provided through pitch and roll DASE channels using rates, attitudes and doppler corrected inertial velocities from the HARS. HAS is used to reduce pilot workload by assisting the pilot in maintaining a desired hover position. HAS is engaged below 15 knots ground speed and 50 KTAS with trim feel system ON, ATTD/Hover Hold switch ON, and the pitch and roll DASE channels are engaged. Additionally, a heading hold mode is provided through the yaw DASE channel using aircraft heading information from the HARS. This function is engaged whenever the yaw DASE channel and HAS is engaged. Schematic diagrams are shown in figures 18 through 20.

16. A limited authority attitude hold mode is provided through pitch and roll SAS. Attitude Hold is engageable above 60 KTAS when the ATTD/Hover Hold switch is on, trim feel system ON, and pitch and roll SAS are engaged. Attitude Hold will automatically disengage whenever the airspeed is decreased to 50 KTAS. Schematic diagrams are shown in figures 20 through 22.

17. 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. A schematic diagram is shown in figure 23.

HYDRAULIC SYSTEM

General

18. The hydraulic system consists of four hydraulic servoactuators powered simultaneously by two independent 3000-psi hydraulic

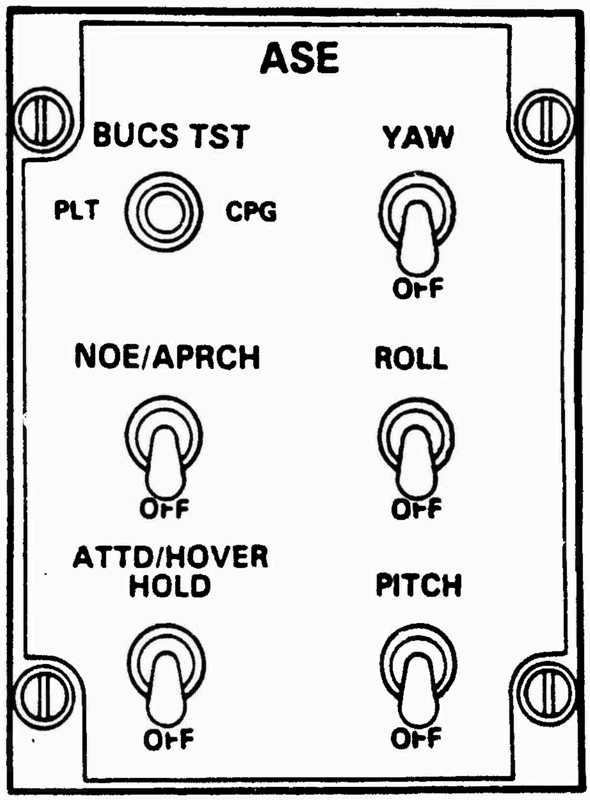


Figure 12. Pilots DASE Control Panel

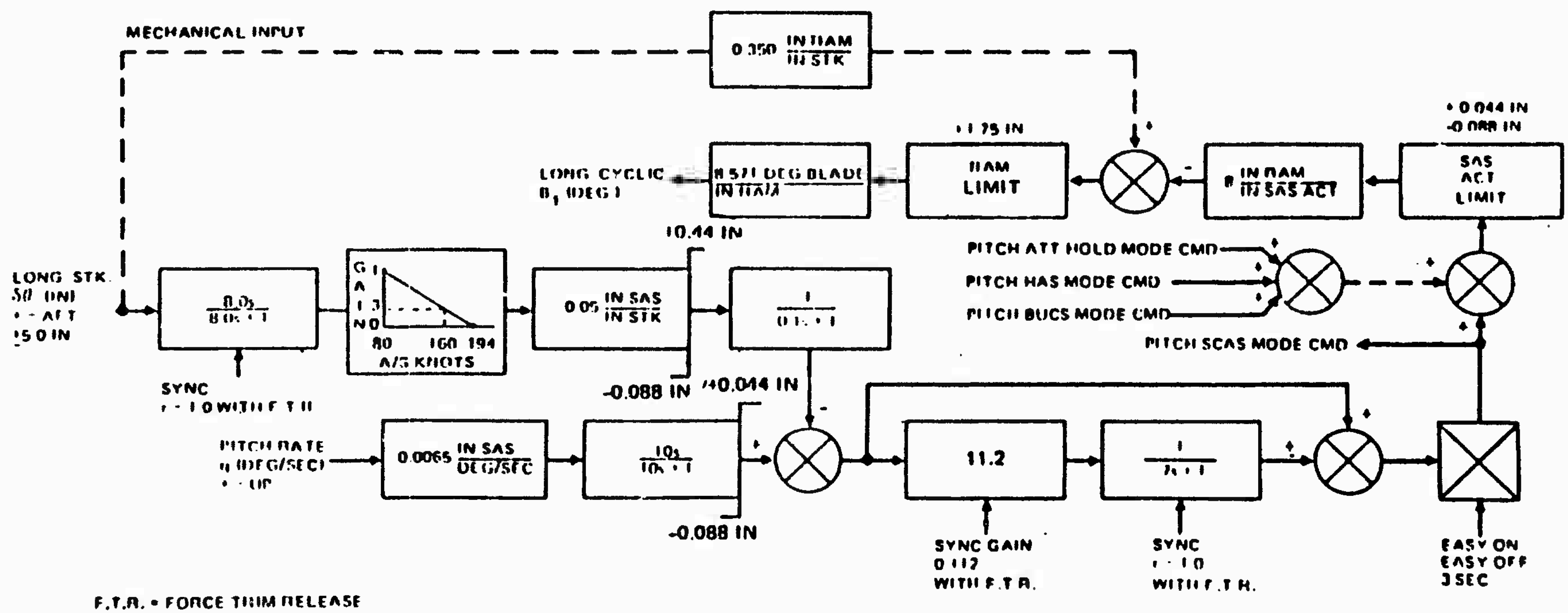


Figure 13. Pitch SCAS Block Diagram

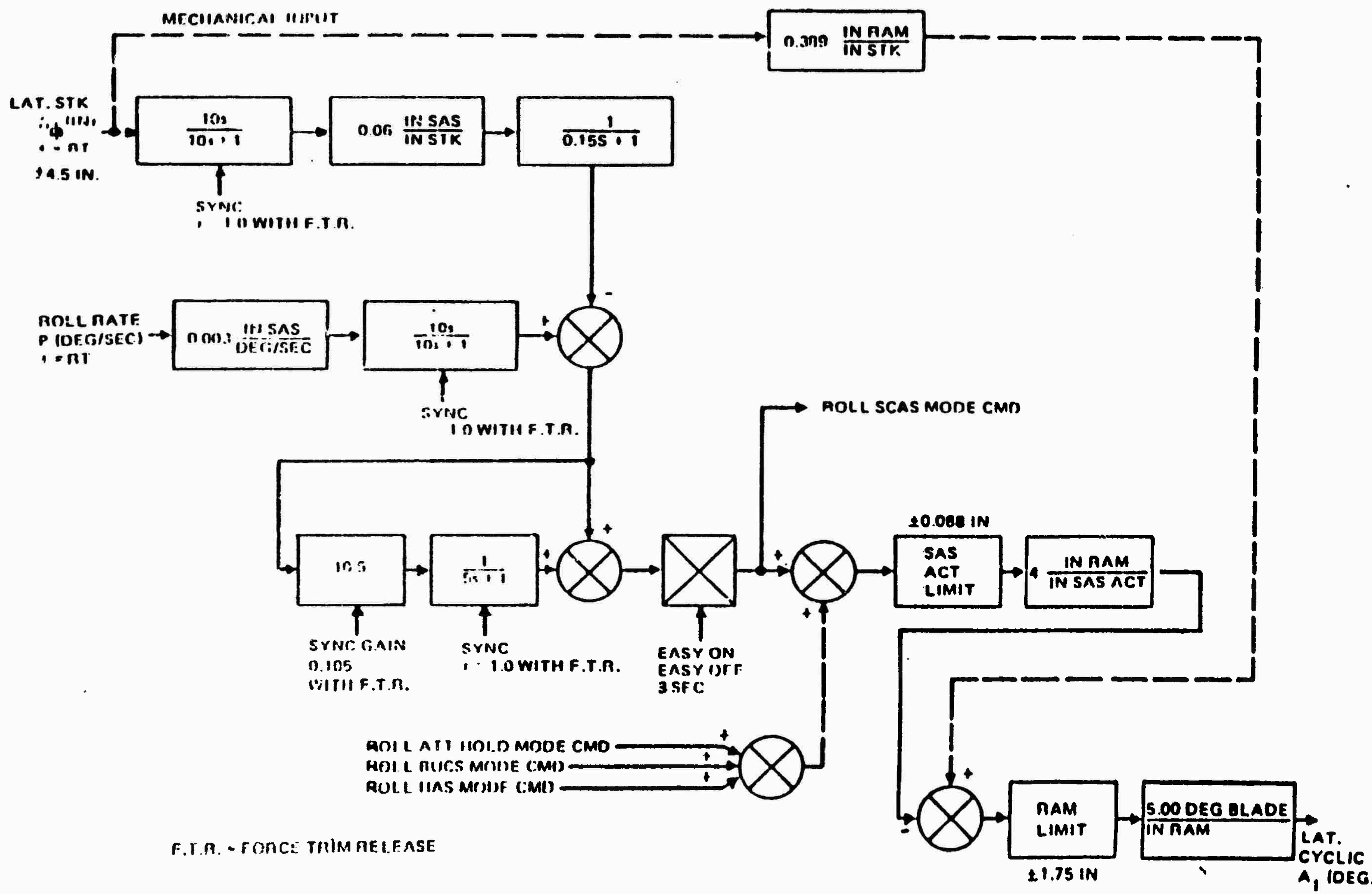


Figure 14. Roll SCAS Block Diagram

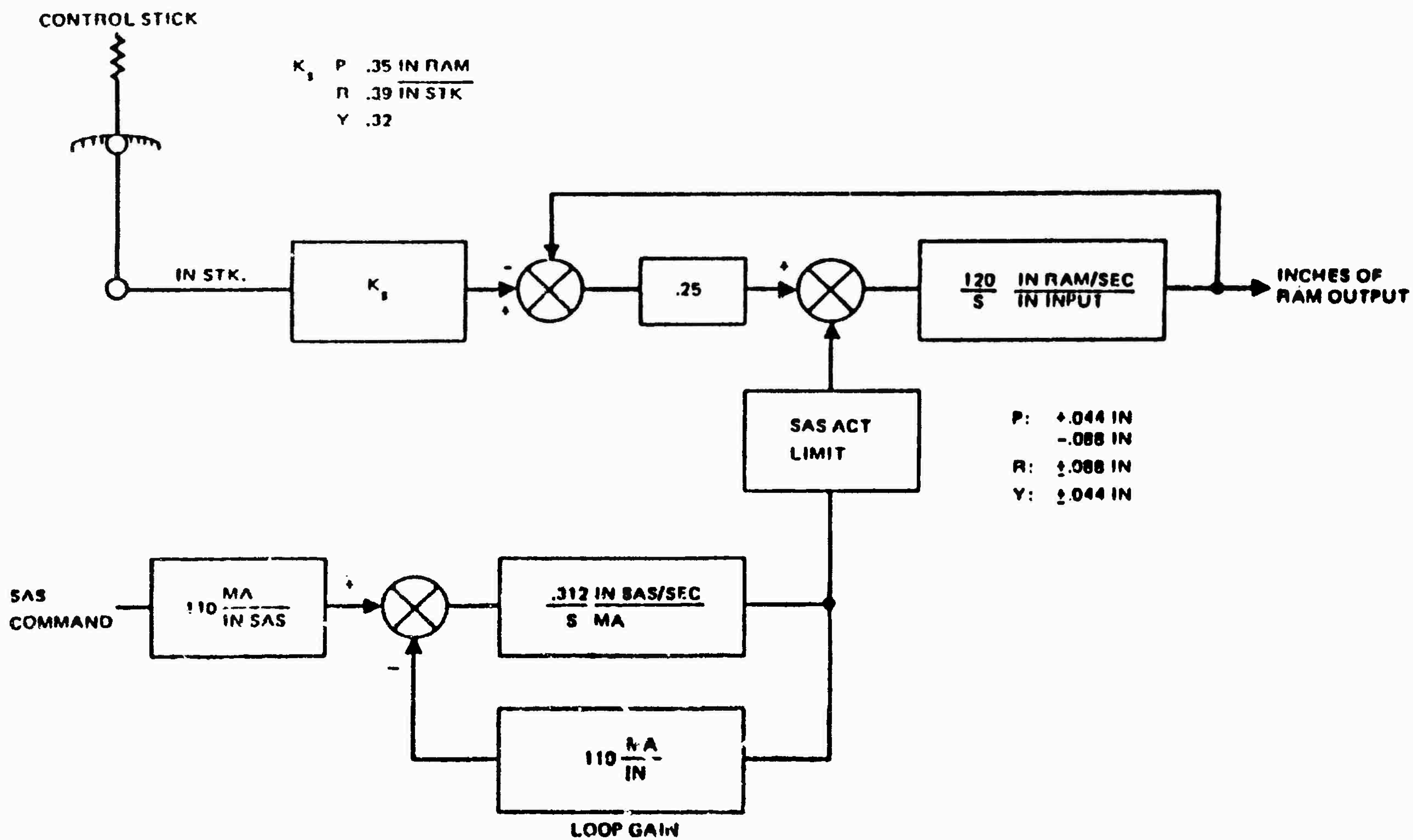


Figure 15. Mechanical and SAS Servo Loop

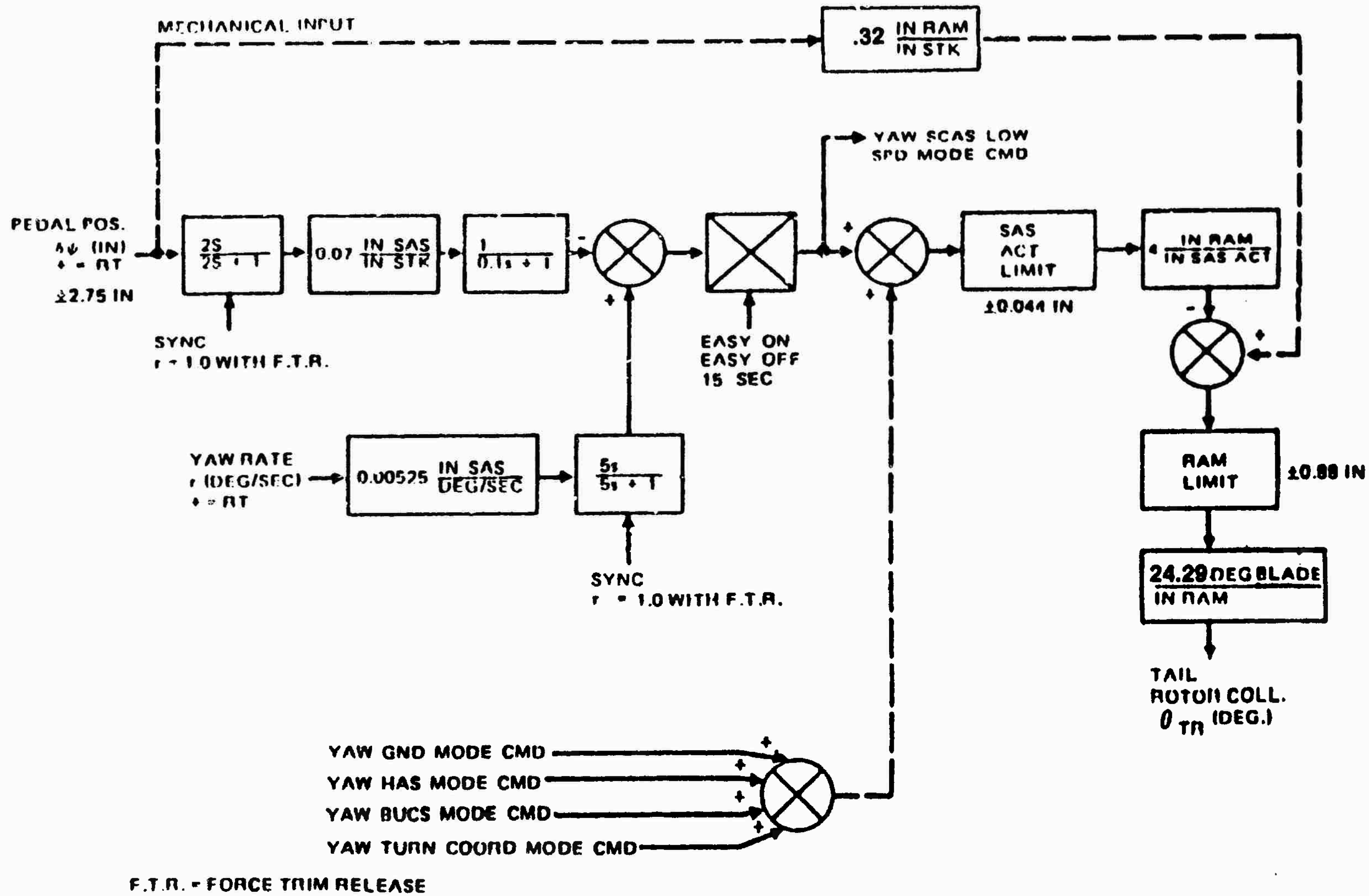


Figure 16. Yaw SCAS Low Speed Mode Block Diagram

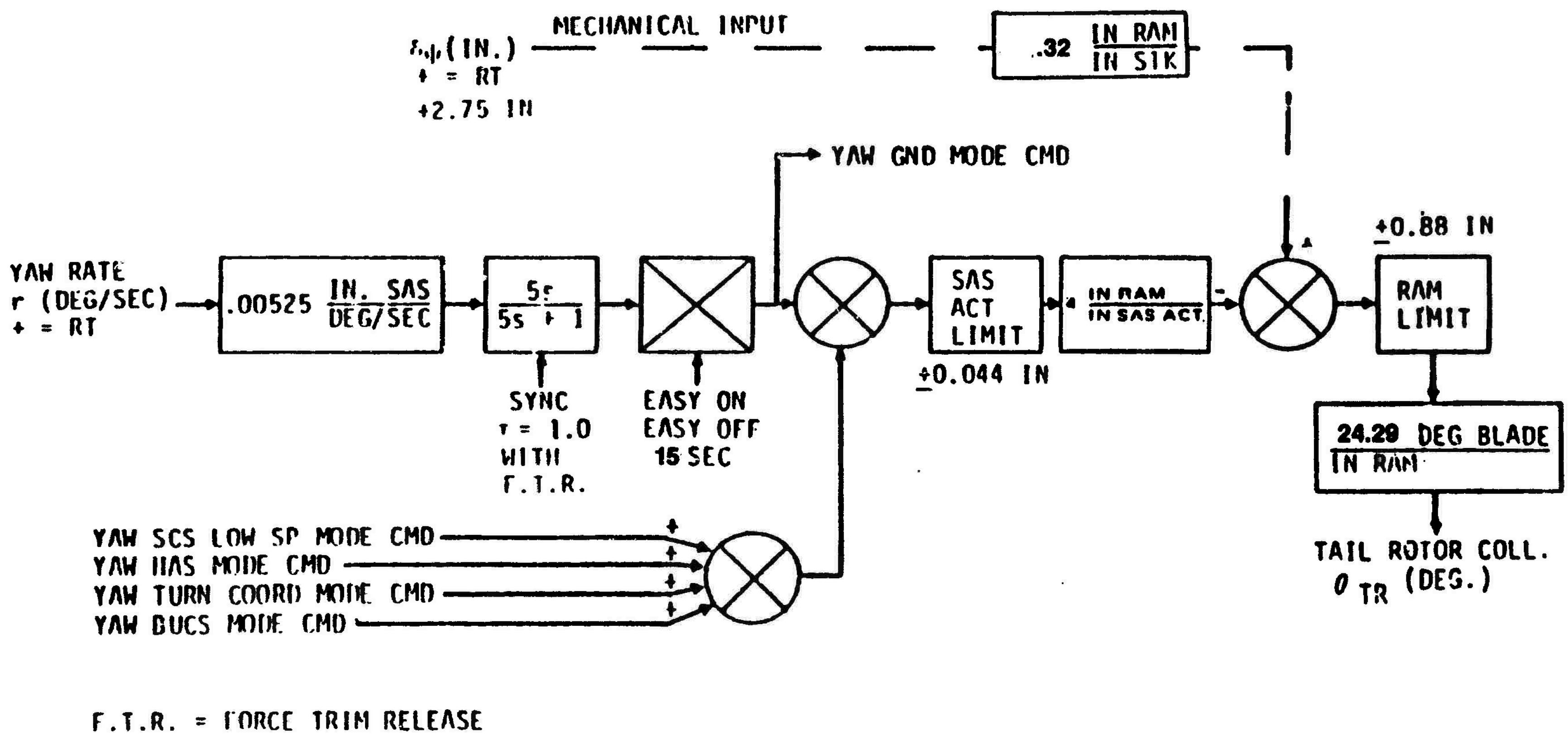


Figure 17. Yaw SCAS Ground Mode Block Diagram

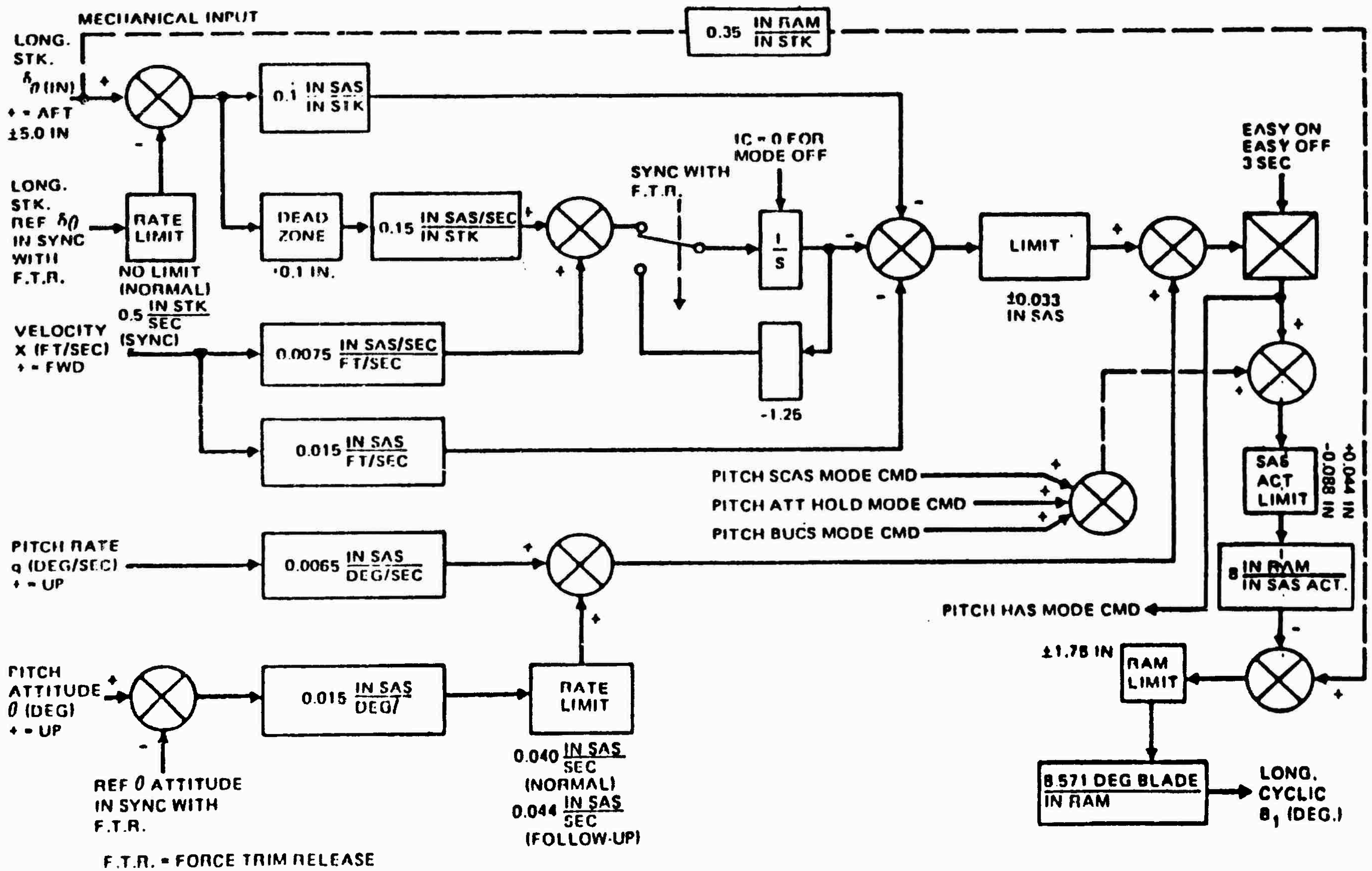


Figure 18. Pitch Hover Augmentation Mode Block Diagram

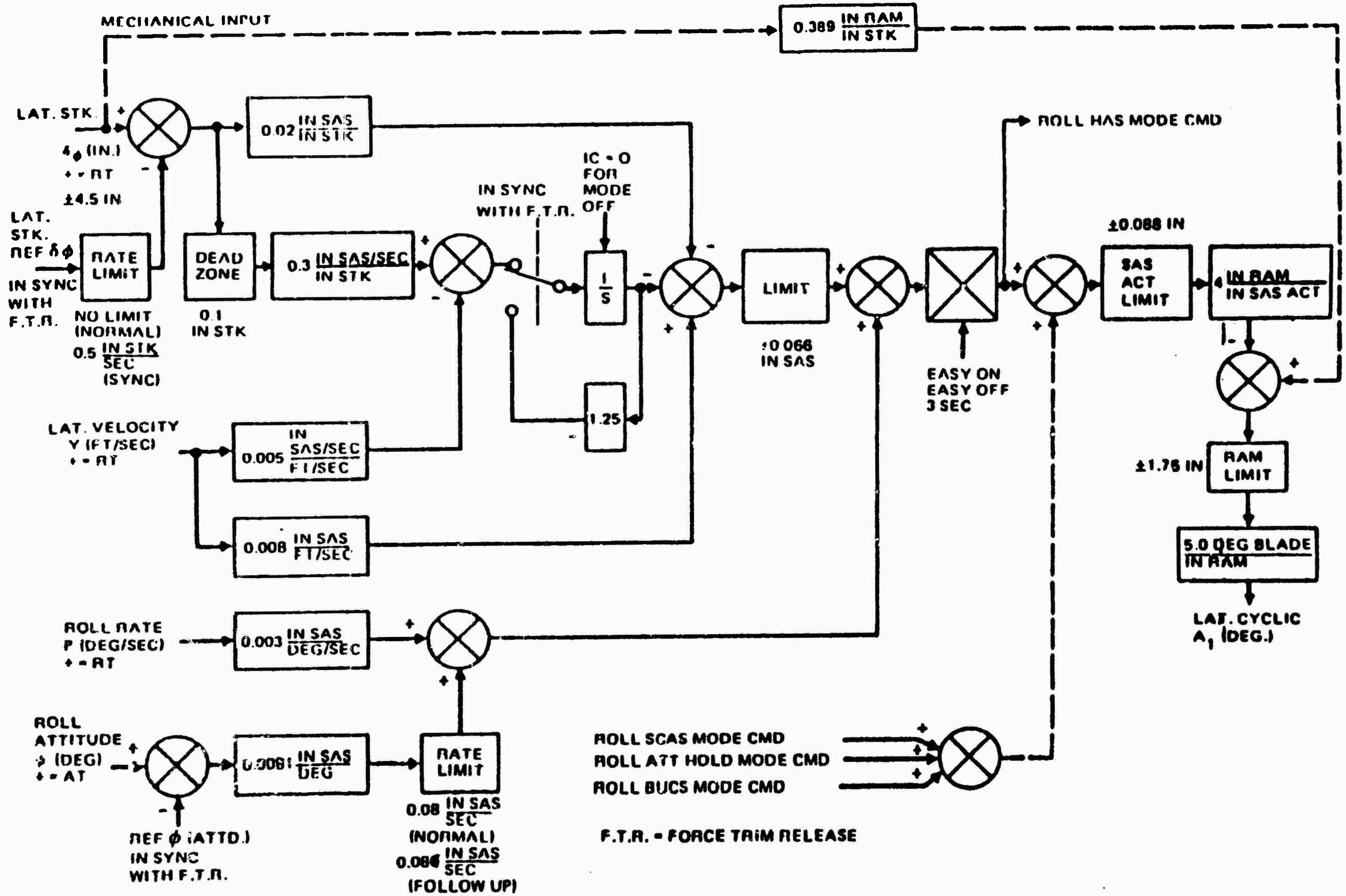


Figure 19. Roll Hover Augmentation Mode Block Diagram

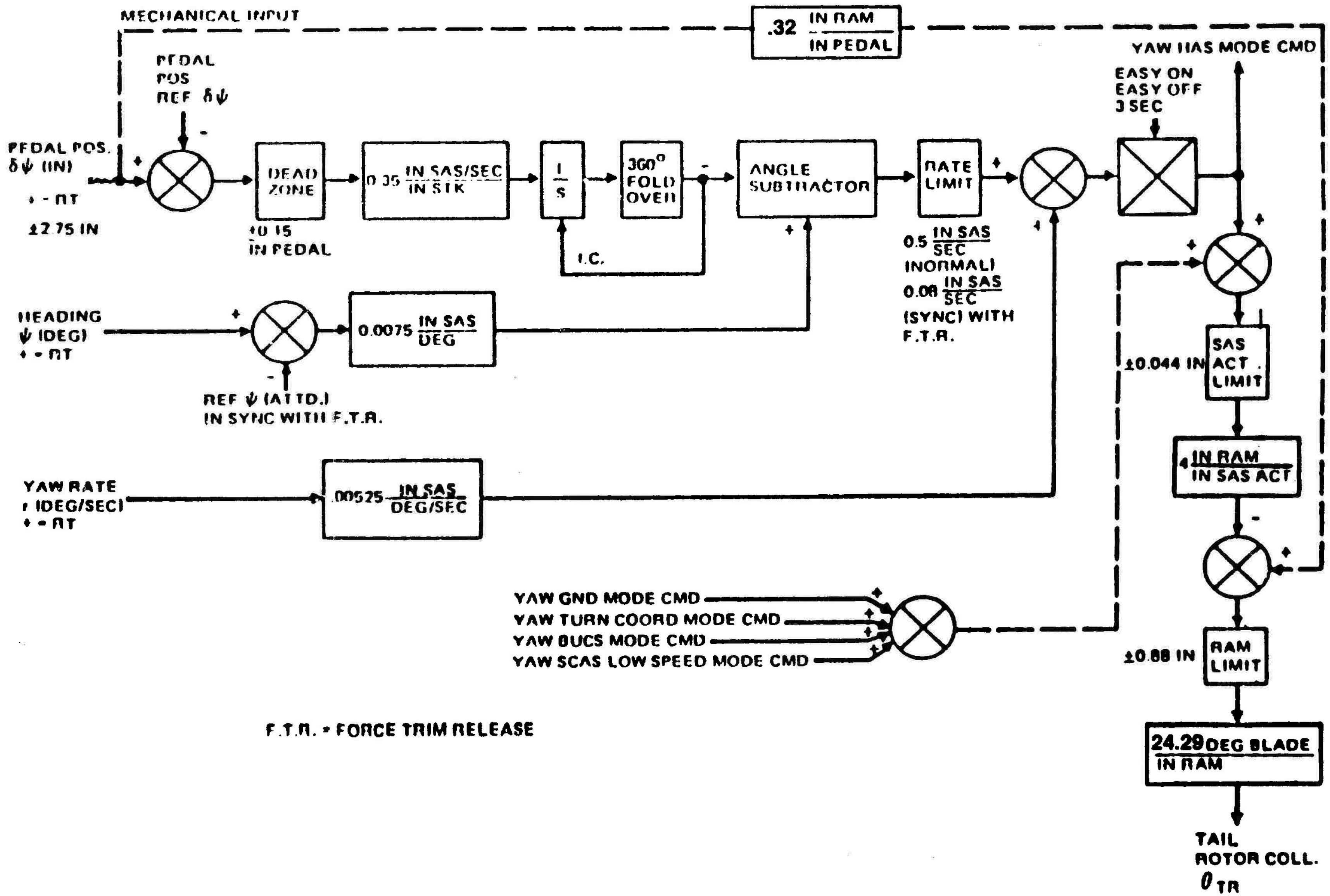


Figure 20. Heading Hold Mode Block Diagram

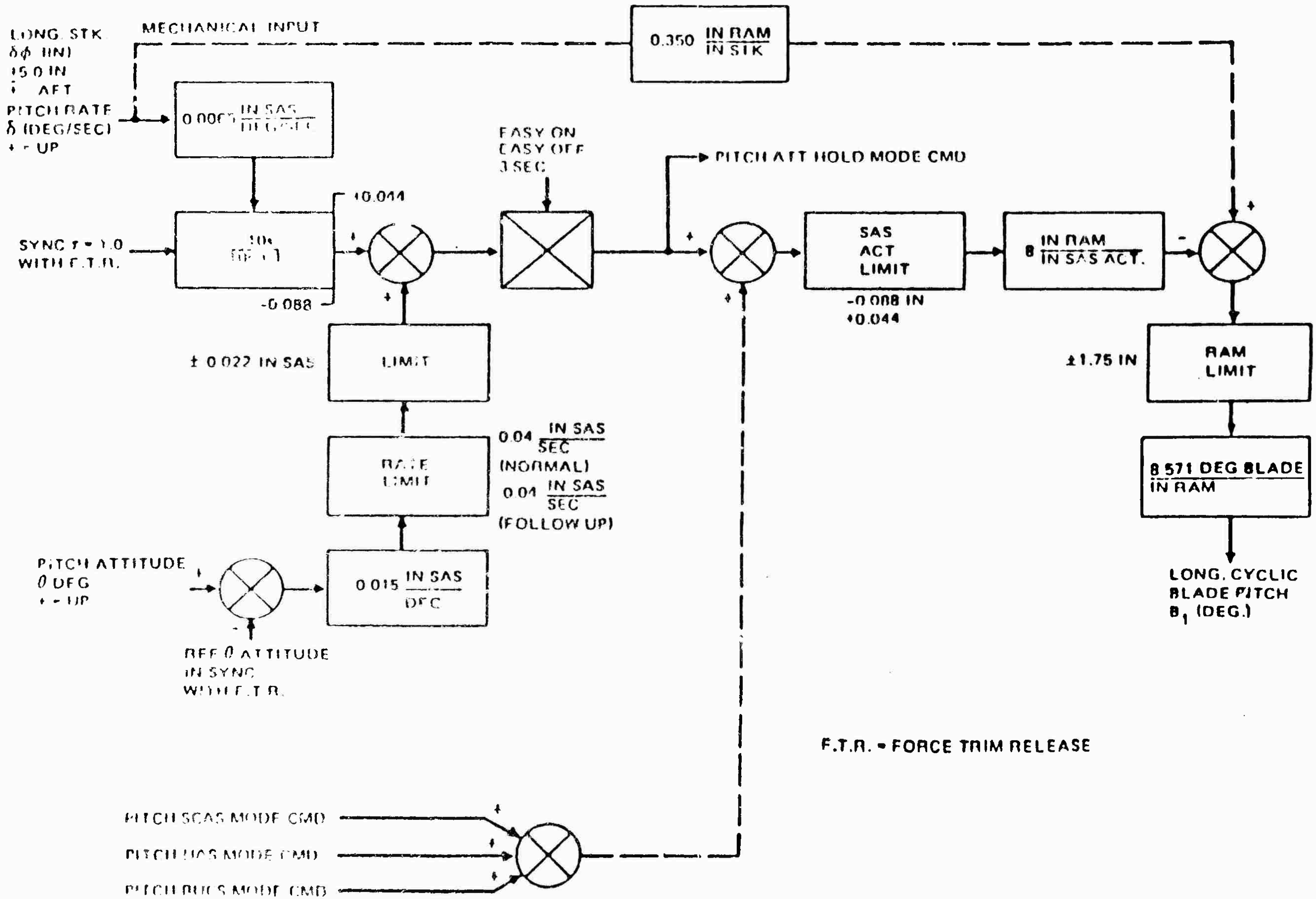


Figure 21. Pitch Attitude Hold Mode Block Diagram

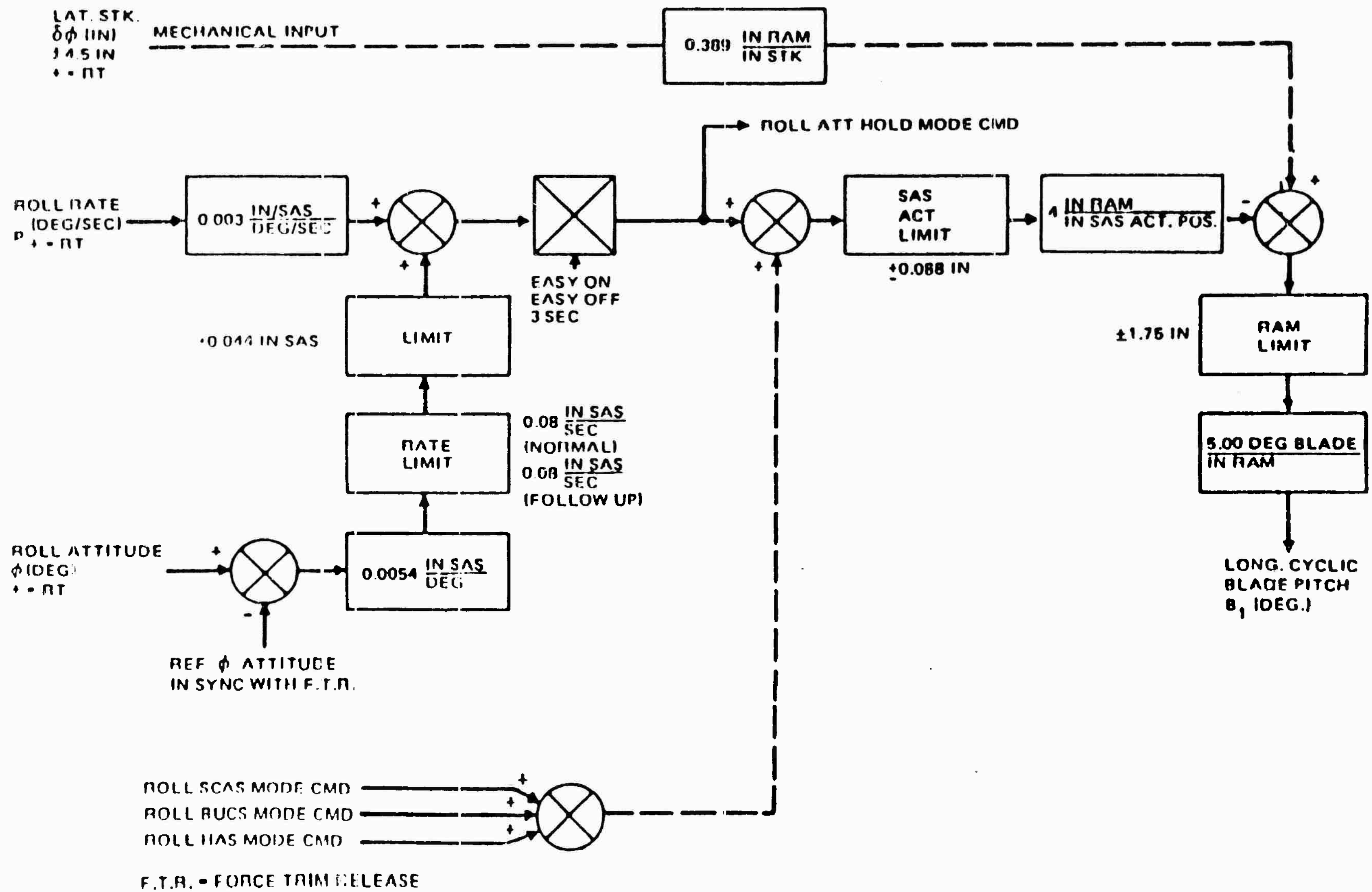


Figure 22. Roll Attitude Hold Mode Block Diagram

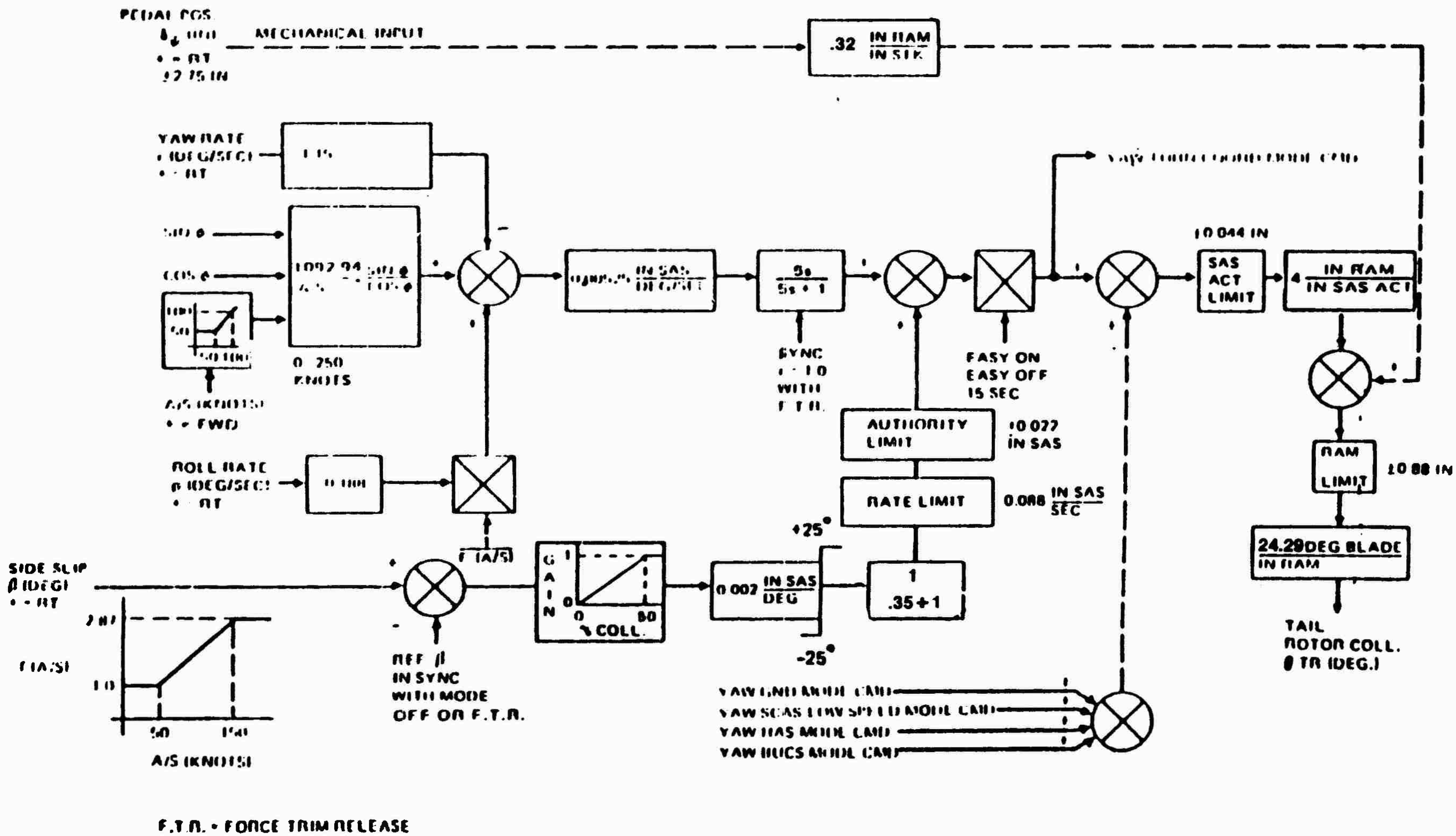


Figure 23. Yaw SCAS Turn Coordination Block Diagram

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 rotors are not turning. 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

20. The primary hydraulic system (fig. 24) 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 serovactuators. The primary system also provides the hydraulic pressure for operation of the DASE functions.

Utility Hydraulic System

21. The utility hydraulic system (fig. 25) 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 or CPG 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.

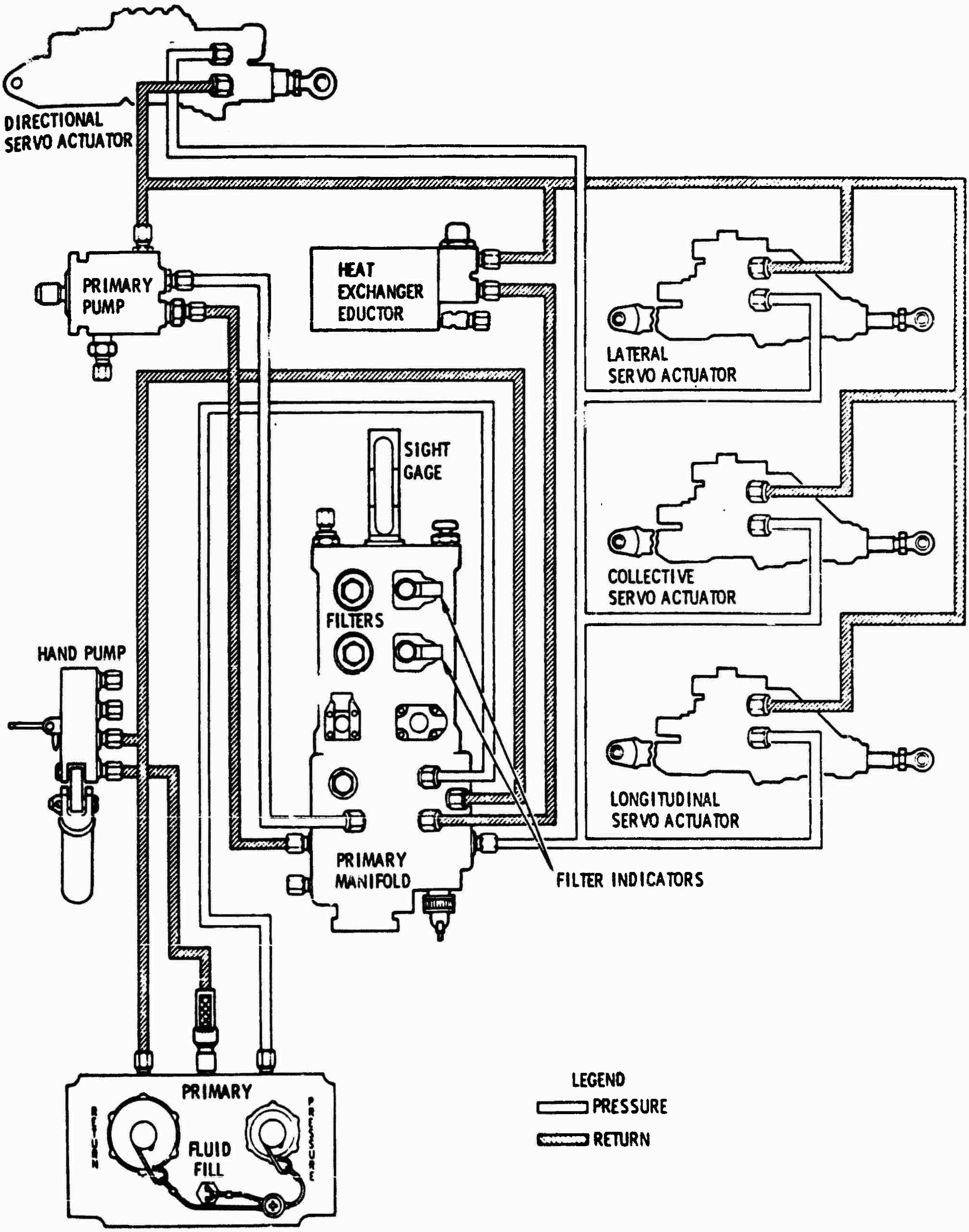


Figure 24. Primary Hydraulic System

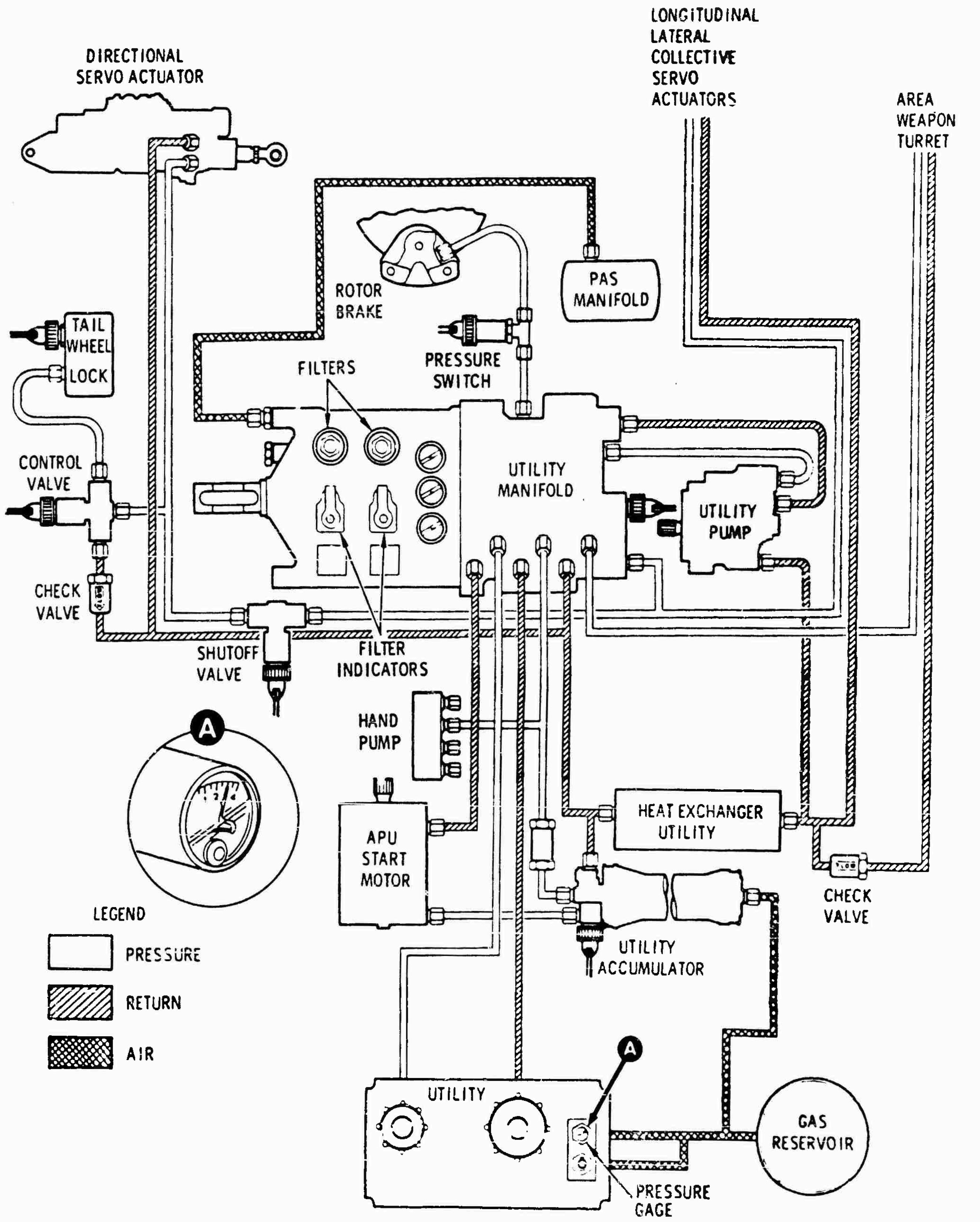


Figure 25. Utility Hydraulic System

Servoactuators

22. Individual hydraulic servoactuators are provided for longitudinal, lateral, collective, and directional controls. Each servoactuator (fig. 26) consists of a ballistically tolerant housing, a single actuator rod and dual frangible pistons, a 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, some control authority will be lost in the directional servoactuator system. DASE and BUCS functions would be lost with failure of the primary system.

ENGINES

23. The AH-64A helicopter, for this test, was powered by two General Electric T700-GE-701 front drive turboshaft engines, rated at 1690 shaft horsepower (shp) at 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 T700-GE-701 engine also incorporates a history recorder which records total engine events. Engines S/N 374115 and 374114 were installed in the left and right positions, respectively. Both engines were equipped with 302J Electrical Control Units (ECUs). ECUs S/N CAOE2019 and CAOE2066 were installed on the left and right engine, respectively. The following engine data are provided:

Model	T700-GE-701
Type	Turboshaft
Rated power (intermediate)	1690 shp sea level, standard day, uninstalled
Output speed (at 100 percent N_R)	20,952 RPM
Compressor	5 axial stages, 1 centrifugal stage

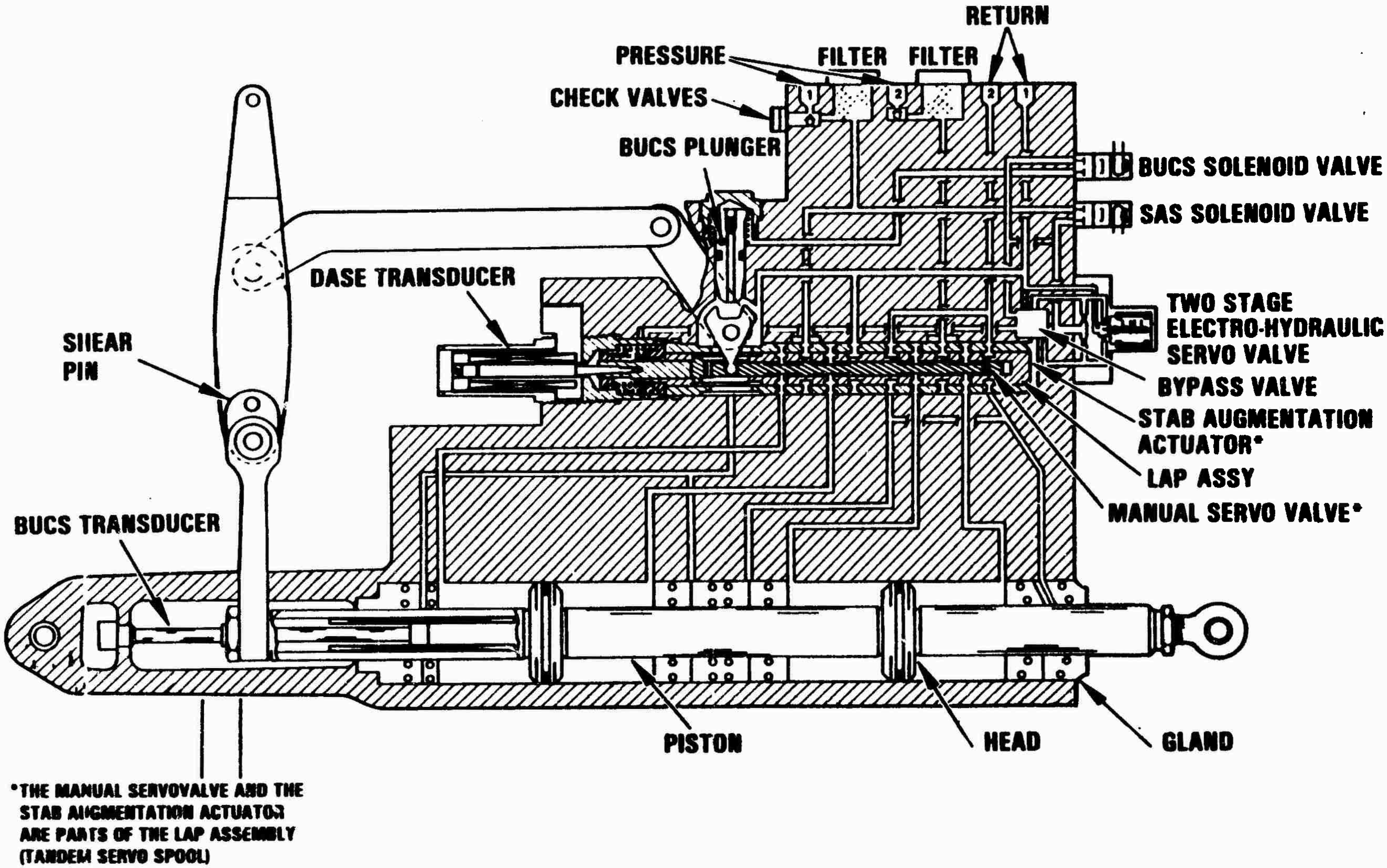


Figure 26. Servoactuators

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
Weight (dry)	423 lb
Length	47 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

24. The IR suppression system 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. 27). 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

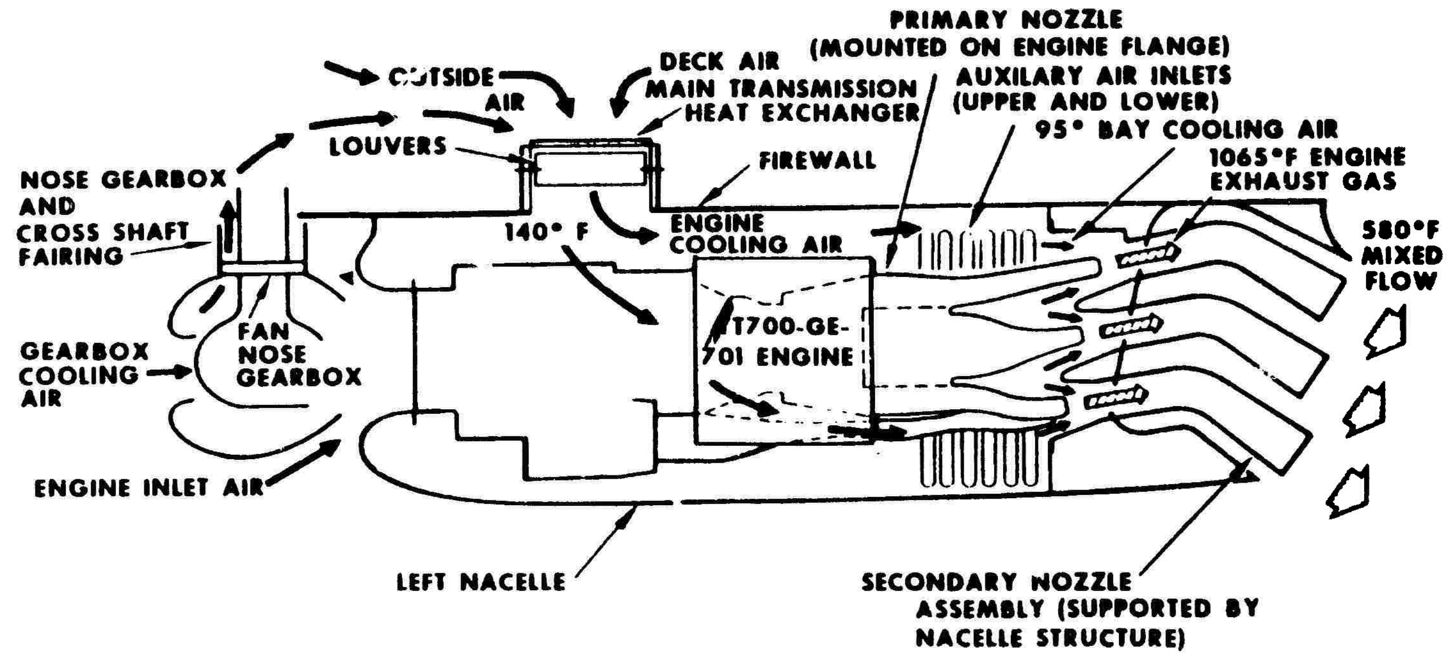


Figure 27. Infrared Suppression System Engine Cooling

provide convective cooling to the engine during shutdown. The movable bottom door is closed by engine bleed air during engine operation.

FUEL SYSTEM

25. The AH-64A 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 28 is a schematic of the fuel system. Figure 29 shows the locations and capacities of the two internal fuel cells.

26. By using the crossfeed (pilot) or tank select (CPG) switch on the fuel control panel (fig. 30), the pilot or CPG can select either or both tanks from which the engines will draw fuel. With the crossfeed or tank select switch in the NORM 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. When FWD tank (pilot) or FROM FWD (CPG) is selected the two fuel crossfeed/shutoff valves are positioned so that both engines draw fuel from the forward tank. The AFT tank (pilot) or FROM AFT (CPG) position allows the engines to draw fuel from the aft tank only. 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 only when the tank select switch (CPG) or crossfeed switch (pilot) is in the aft position.

27. The pilot and CPG also have the capability to transfer fuel between tanks using the transfer switch on the fuel control panels. Moving the fuel transfer switch out of the OFF position closes the refuel valve and activates the air-driven pump which transfers fuel in the selected direction.

28. Fuel is supplied to the APU from the AFT fuel cell through the APU shutoff valve and APU boost pump. The APU shutoff valve is controlled by the APU off-run-start switch located on the APU control panel (fig. 31). When this switch is placed to the run position the shutoff valve is electrically opened and fuel is supplied to the APU boost pump.

29. External refueling provisions are provided on the right side of the aircraft. The forward and aft cells may be gravity filled separately. The single point adapter or closed circuit adapter allow the fore and aft cells to be pressure refueled separately or

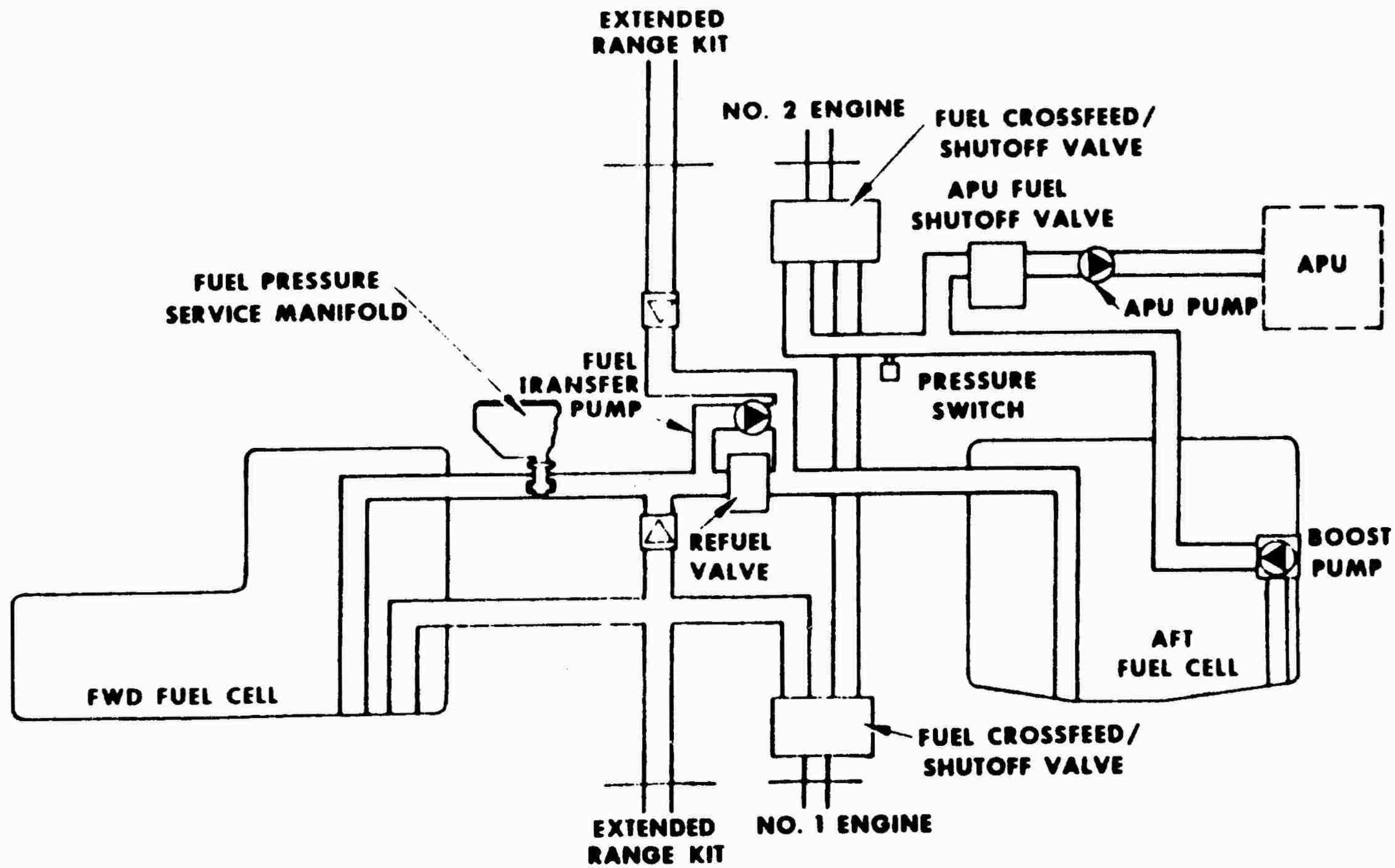


Figure 28. Fuel System Major Components

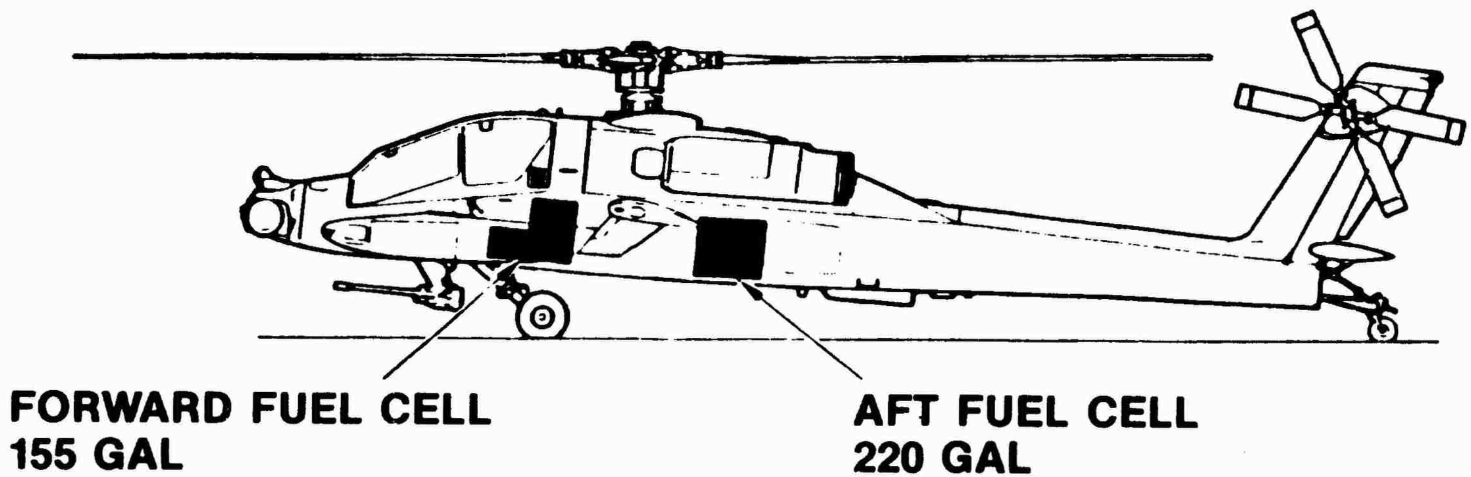


Figure 29. Fuel Cell Locations

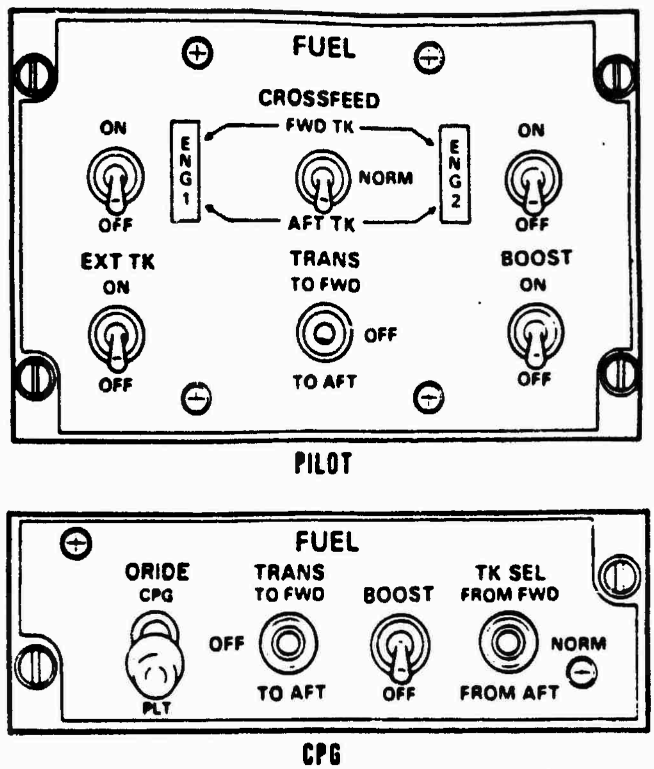
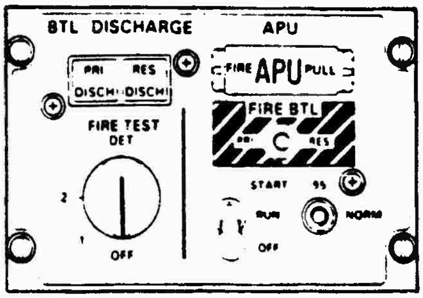


Figure 30. Pilot and Copilot/Gunner Fuel Control Panels



APU CONTROL PANEL

Figure 31. APU Control Panel

simultaneously using the external refueling panel shown in figure 32.

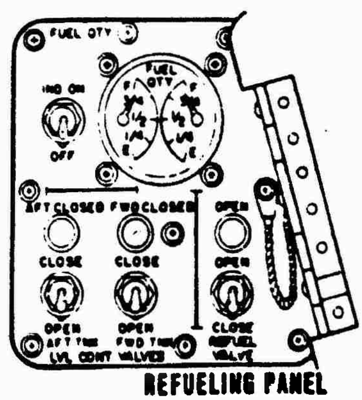


Figure 32. External Refueling Panel

APPENDIX C. INSTRUMENTATION

1. An airborne data acquisition system was installed and maintained by Hughes. The system utilized pulse code modulation (PCM) encoding. Magnetic tape was used to record parameters aboard the aircraft. The various external instrumentation drag items are shown in photos 1 through 7.

2. A boom was mounted on 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 (photo 1).

3. Instrumentation and related special equipment installed are presented in the following lists. The pilot and copilot-gunner stations are shown in photos 8 and 9.

<u>Pilot Station (aft cockpit displays)</u>

Pressure altitude (boom)
Airspeed (boom)
Vertical rate of climb
Main rotor speed
Engine torque (both engines)
Engine measured gas temperature (both engines)
Engine power turbine speed (both engines)
Engine gas producer speed (both engines)
Angle of sideslip
Event switch
Radar altitude
Control Positions
    Longitudinal
    Lateral
    Directional
    Collective
Stabilator incidence angle
Normal acceleration (cg)
Horizontal situation indicator with Doppler interface
Primary attitude reference
Turn needle and ball
Sensitive cg lateral acceleration indicator

<u>Copilot Station Displays</u>

Airspeed (ship)
Altitude (ship)
Main rotor speed
Engine torque (both engines)
Engine measured gas temperature (both engines)
Engine gas producer speed (both engines)

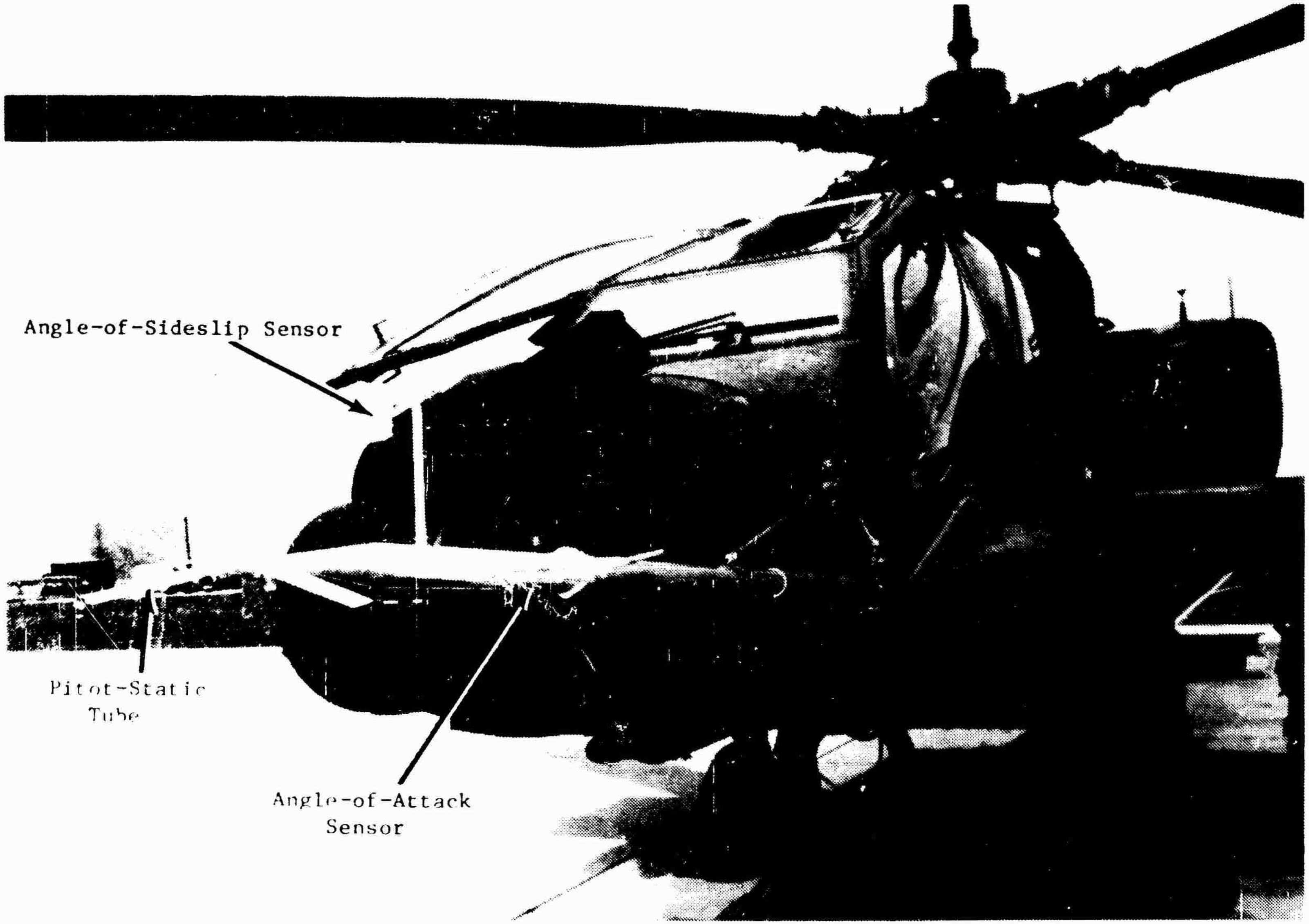


Photo 1. Instrumentation Boom

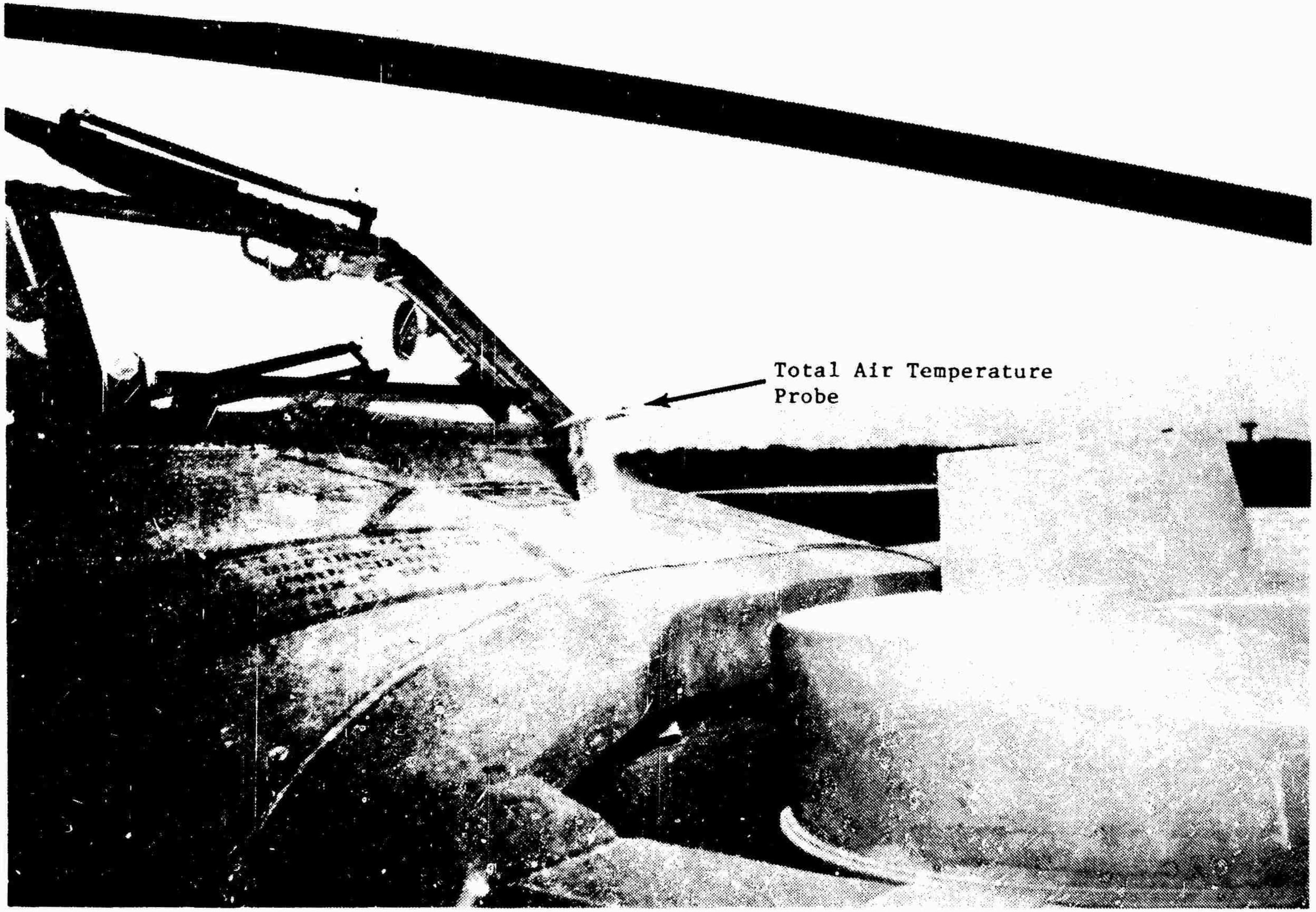


Photo 2. Total Air Temperature Probe

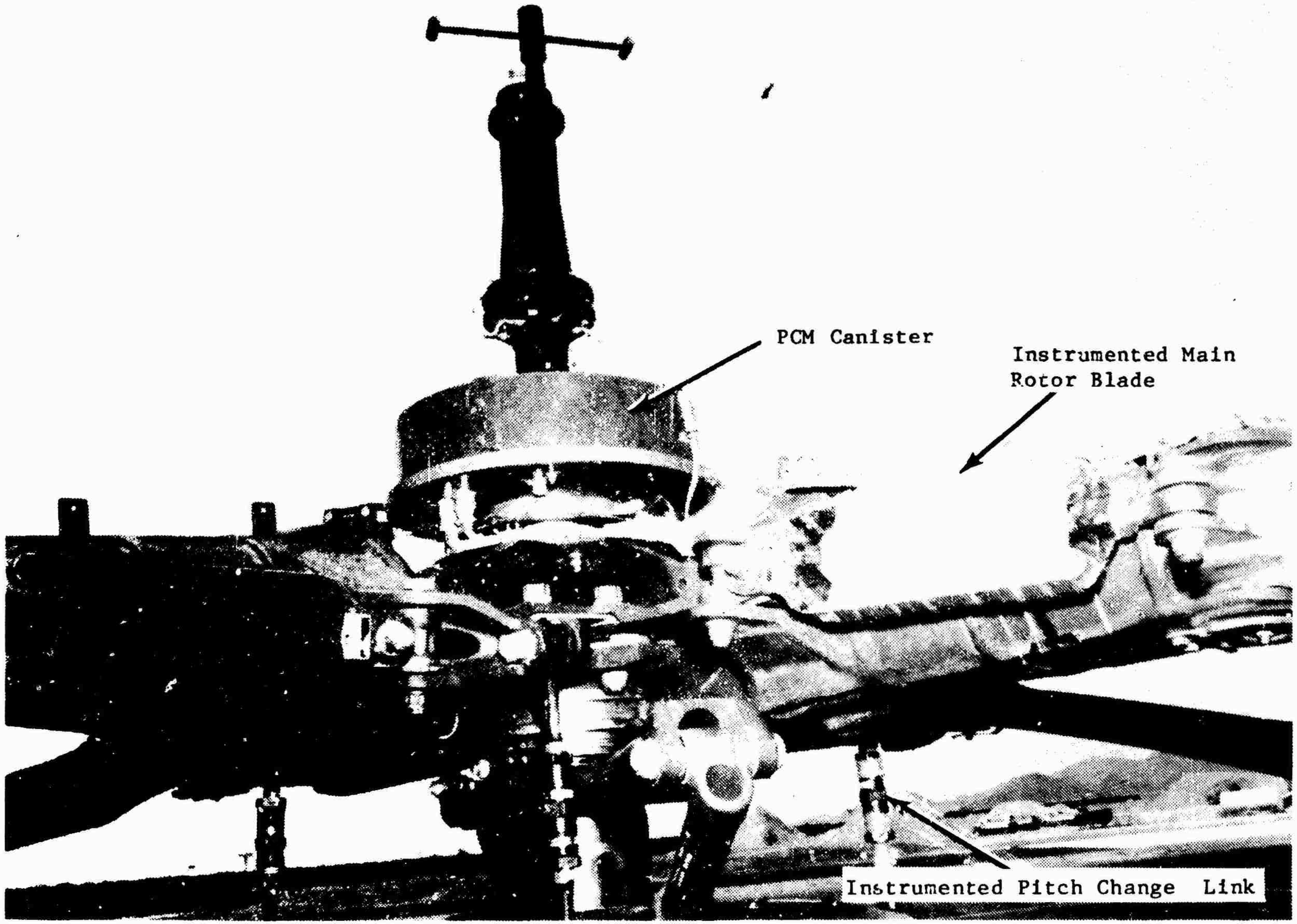


Photo 3. Main Rotor Instrumentation

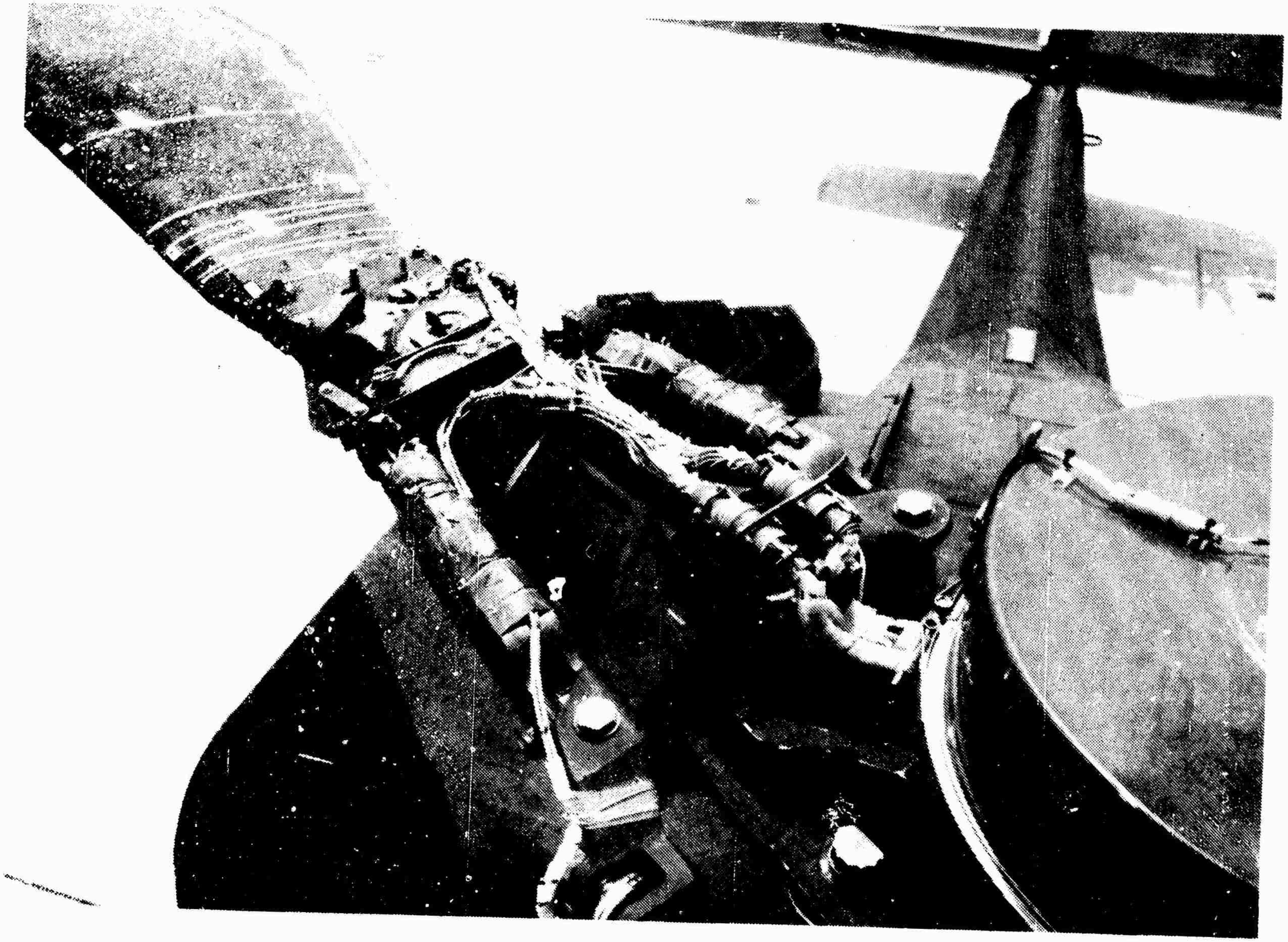


Photo 4. Top View, Instrumented Main Rotor Blade

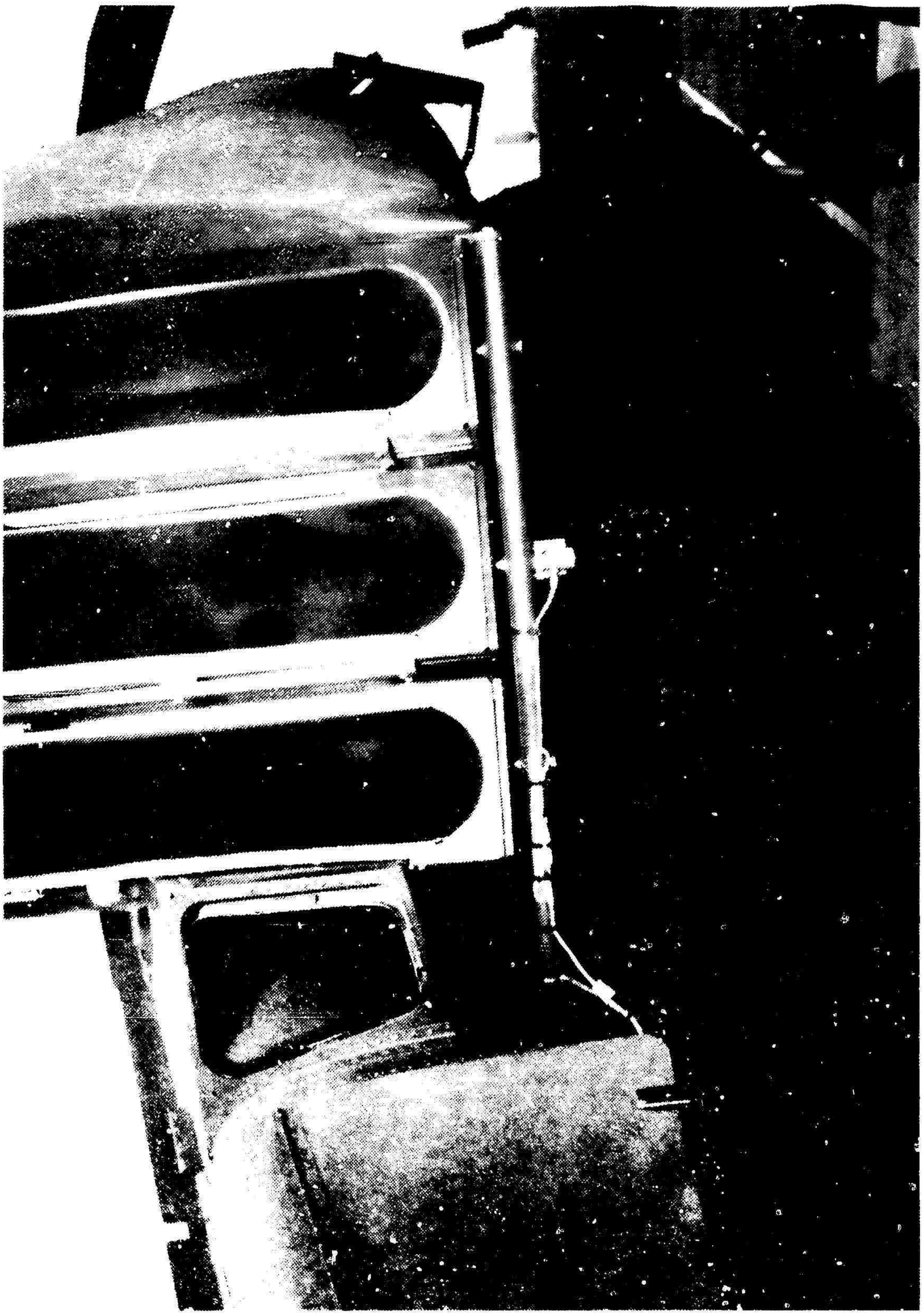


Photo 5. Accelerometer Located on Right Engine Infrared Suppressor

Photo 6. PCM Antenna (Located at Approximately Station 480)

Photo 7. Instrumented Tail Rotor

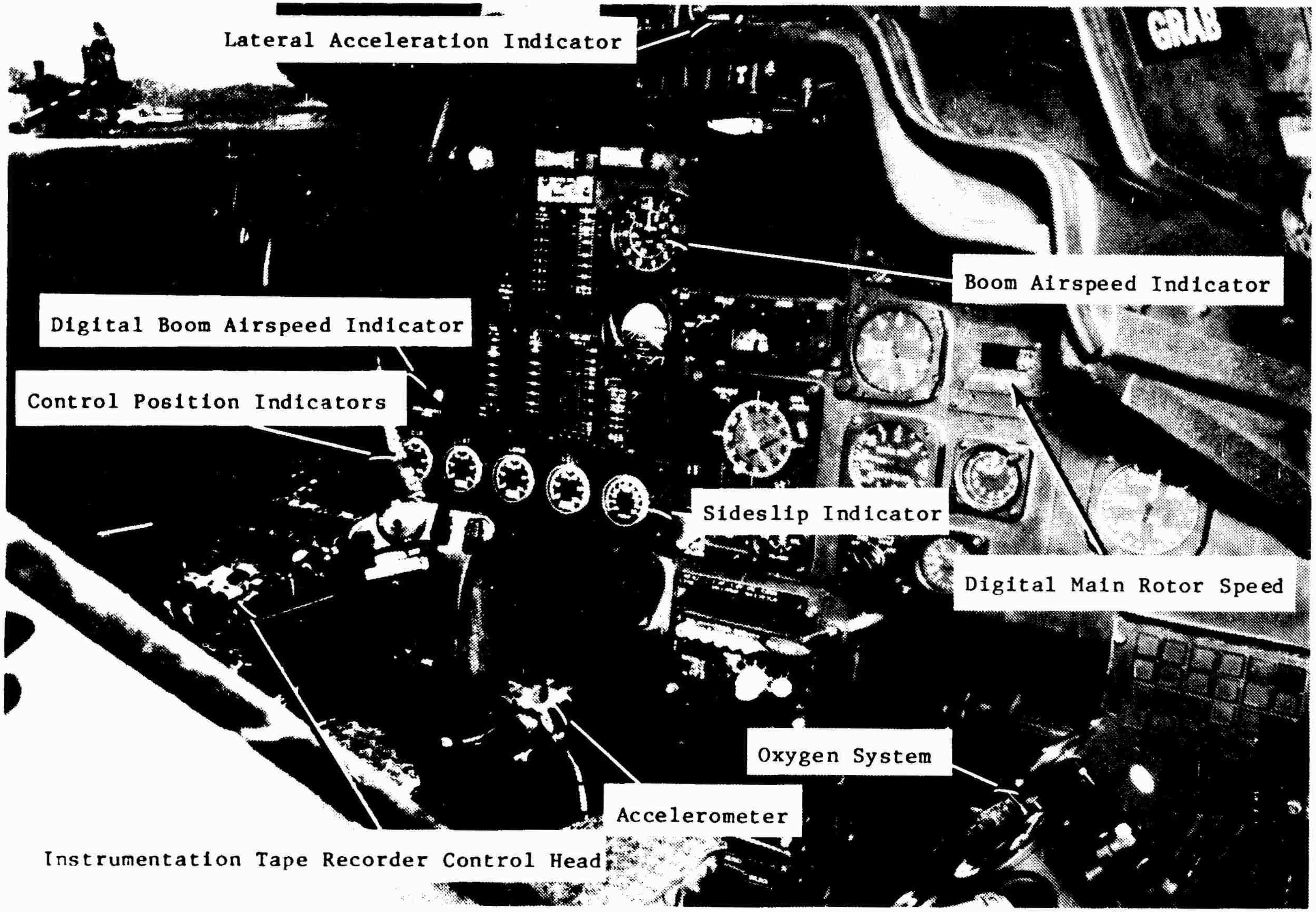


Photo 8. Pilot Station

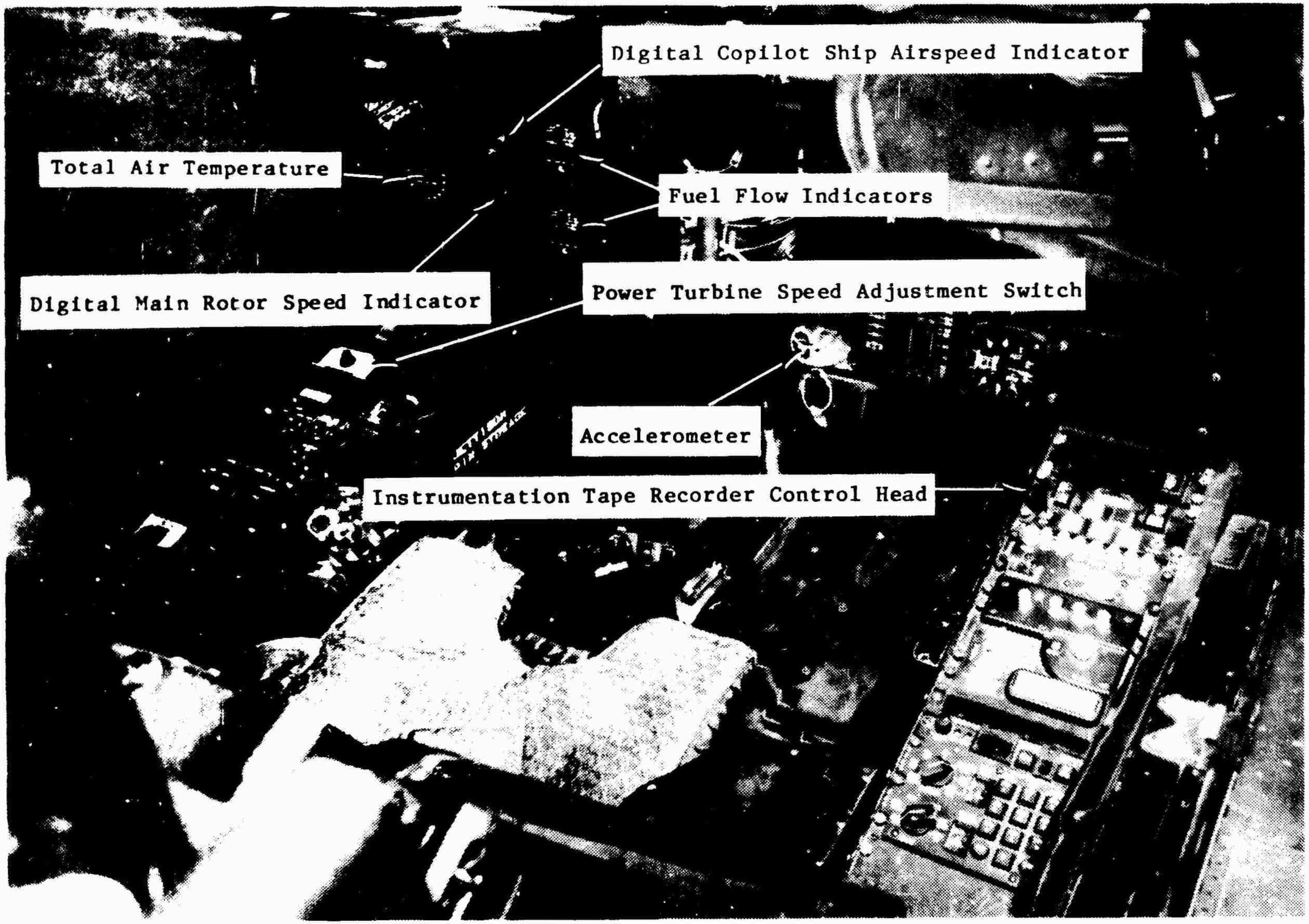


Photo 9. Copilot-Gunner Station

Fuel used (both engines)
Fuel used (APU)
Total air temperature
Time code display
Event switch
Data system controls
Doppler

PCM Parameters (Magnetic Tape)

Time code
Event
Main rotor speed
Fuel temperature (both engines)
Fuel used (both engines)
Engine fuel flow rate (both engines)
Fuel used (APU)
Engine torque (both engines)
Engine measured gas temperature (both engines)
Engine gas producer speed (both engines)
Engine power turbine speed (both engines)
Airspeed (boom)
Airspeed (ship, left and right)
Altitude (boom)
Altitude (ship, left and right)
Total air temperature (boom)
Angle of attack (boom)
Angle of sideslip (boom)
Control positions
- Longitudinal cyclic
- Lateral cyclic
- Pedal
- Collective

Stabilator incidence angle
Aircraft attitudes
- Pitch
- Roll
- Yaw

Aircraft angular velocities
- Pitch
- Roll
- Yaw

Vibration Accelerometers
- Pilot seat (3 axes)
- Copilot seat (3 axes)
- Aircraft cg (3 axes)

Stability augmentation system actuator positions
- Longitudinal

Lateral
Directional
Control actuator positions
Tail rotor
Collective pitch
Cyclic pitch
Cyclic roll
Air data system
Longitudinal velocity
Lateral velocity
Pressure altitude
Outside air temperature
Angle of sideslip
Resultant airspeed
Radar altitude
CG normal acceleration
CG lateral acceleration
Rotor azimuth

4. Position error of the boom airspeed system was determined and is presented in figure 1.

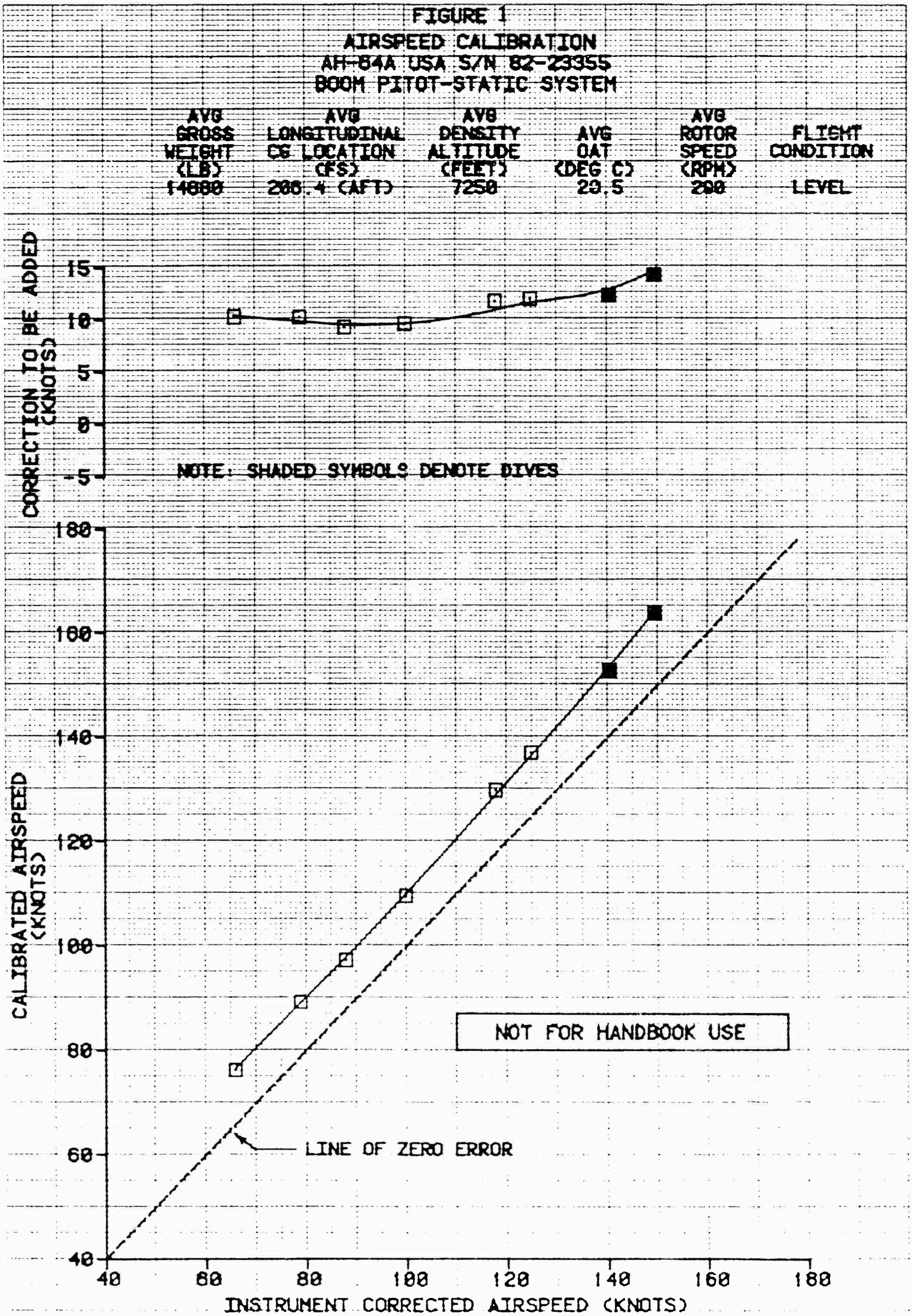


FIGURE 1
AIRSPEED CALIBRATION
AH-64A USA S/N 82-23355
BOOM PITOT-STATIC SYSTEM
AVG GROSS WEIGHT (LB) 14880
AVG LONGITUDINAL CG LOCATION (FS) 206.4 (AFT)
AVG DENSITY ALTITUDE (FEET) 7250
AVG OAT (DEG C) 20.5
AVG ROTOR SPEED (RPM) 290
FLIGHT CONDITION LEVEL
CORRECTION TO BE ADDED (KNOTS)
NOTE: SHADED SYMBOLS DENOTE DIVES
CALIBRATED AIRSPEED (KNOTS)
NOT FOR HANDBOOK USE
LINE OF ZERO ERROR
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. Performance data were obtained using the methods described in Army Materiel Command Pamphlet AMCP 706-204 (ref 14, app A) as a guide. Handling qualities data were evaluated using test methods described in Naval Air Test Center Flight Test Manual FTM No. 101 (ref 15) as a guide.

AIRCRAFT WEIGHT AND BALANCE

2. The aircraft was weighed in the 8-Hellfire configuration, as instrumented for test, with full oil, residual fuel after draining, and ballast required for the test. In this configuration, the aircraft weighed 13,078 pounds with a longitudinal center of gravity (cg) at fuselage station (FS) 209.2. The lateral cg was at butt line (BL) -0.9. The forward fuel tank was limited to a maximum of 900 pounds of fuel by the Airworthiness Release. This restriction caused all tests to be done at an aft cg.

LEVEL FLIGHT PERFORMANCE

General

3. Level flight performance was generalized by the use of non-dimensional coefficients as follows:

a. Coefficient of power (C_p):

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

b. Coefficient of thrust (C_T):

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

c. Advance Ratio (μ)

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

Where:

SHP = Engine output shaft horsepower (both engines)

ρ = Ambient air density ($lb\text{-}sec^2/ft^4$)

A = Main rotor disc area = 1809.56 ft^2

Ω = Main rotor angular velocity = 30.26 radians/sec (at 289 RPM)

R = Main rotor radius = 24.0 ft

GW = Gross weight (lb)

V_T = True airspeed (kt) = $\dfrac{V_C}{\sqrt{\rho/\rho_0}}$

1.6878 = Conversion factor (ft/sec)/kt

ρ_0 = 0.0023769 ($lb\text{-}sec^2/ft^4$)

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

$\Omega R = 726.34$

$A(\Omega R)^2 = 9.54657879 \times 10^8 \ ft^4/sec^2$

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

4. Specific range (NAMPP) data were calculated from test data using the following equation:

$$NAMPP = \frac{V_T}{W_f} \qquad (4)$$

where:

NAMPP = Nautical airmiles per pound of fuel

V_T = True airspeed (kt)

W_f = Fuel flow (lb/hr)

Corrections to Test Data

5. No corrections were made to the test data presented in this report. Possible sources of error were: a. the drag of the test airspeed boom and the rotating PCM canister; b. the power consumed by the environmental control unit which was used during the test, and; c. the difference in electrical power consumption between the test instrumentation and the mission equipment which would be used during an operational flight.

Shaft Horsepower Required

6. The engine output shaft torque was determined by use of the engine torque sensor. The power turbine shaft twists an amount proportional to the torque in the shaft. A concentric reference shaft is secured by a pin at the front end of the power turbine drive shaft and is free to rotate relative to the power turbine shaft at the rear end. The relative rotation at the rear end of the two shafts is a measure of the torque being transmitted by the power turbine shaft. The two shafts have reference teeth at the rear end which pass by a magnetic pickup. This pickup sends an electrical pulse to the engine electronic control unit (ECU) each time a reference tooth passes by. From these pulses, the ECU calculates the twist in the shaft and converts that twist to torque. A calibration of the torque measuring system was performed in an engine test cell during the engines' acceptance runs. These data are presented in figures 1 and 2 and were used to correct the indicated torque readings during the performance testing. The output SHP was determined from the engine output shaft torque and rotational speed by the following equation:

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

Where:

Q = Engine output shaft torque (ft-lb)

N_p = Engine output shaft rotational speed (RPM)

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

Test Method

7. Level flight performance was determined by using equations 1, 2, and 3. Each speed-power was flown at a constant thrust

FIGURE 1

ENGINE TORQUEMETER CALIBRATION
T700-GE-701 S/N 374-115

NOTES: 1. POWER TURBINE SPEED = 20952 RPM
2. ENGINE INSTALLED ON LEFT SIDE OF AIRCRAFT

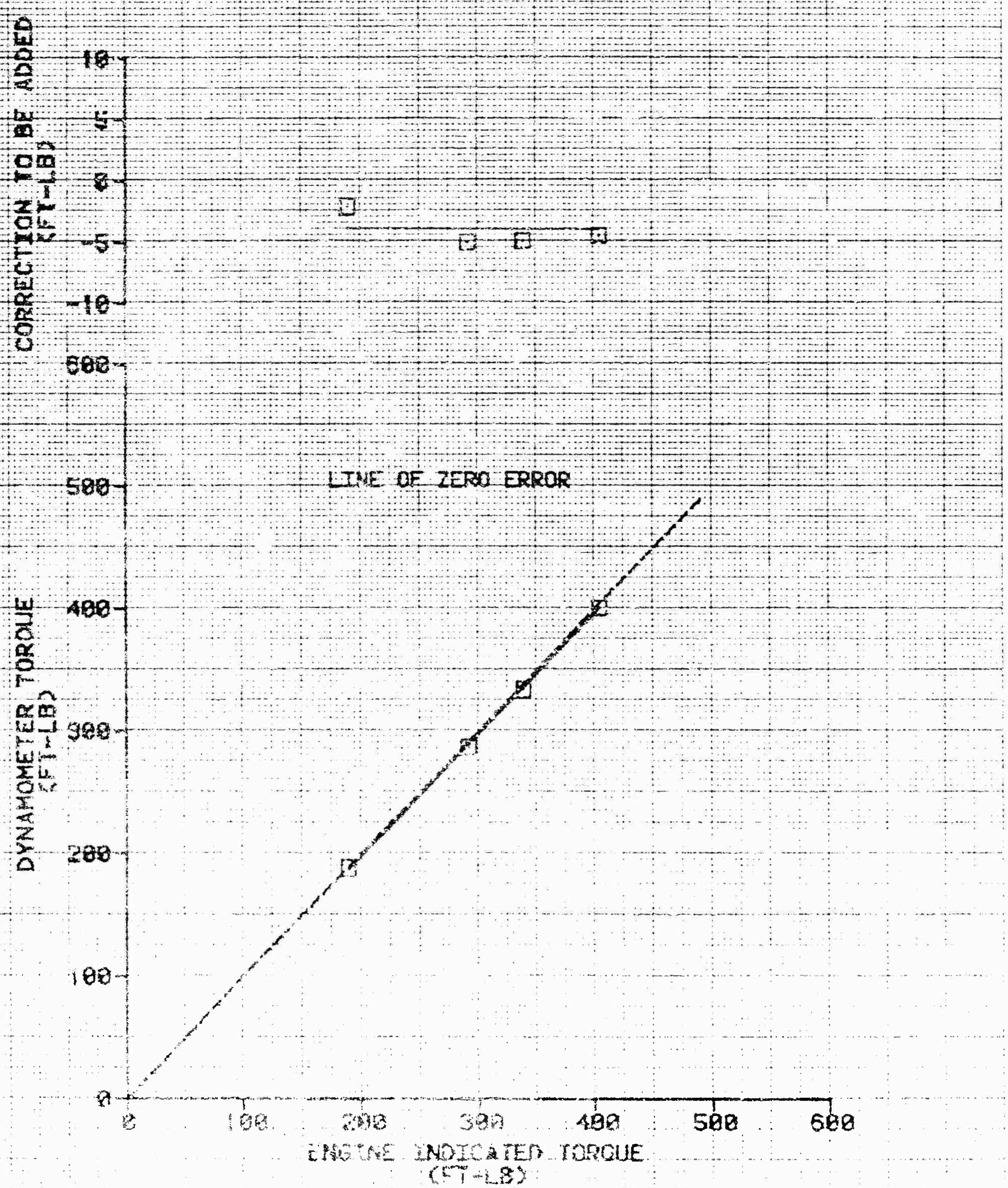
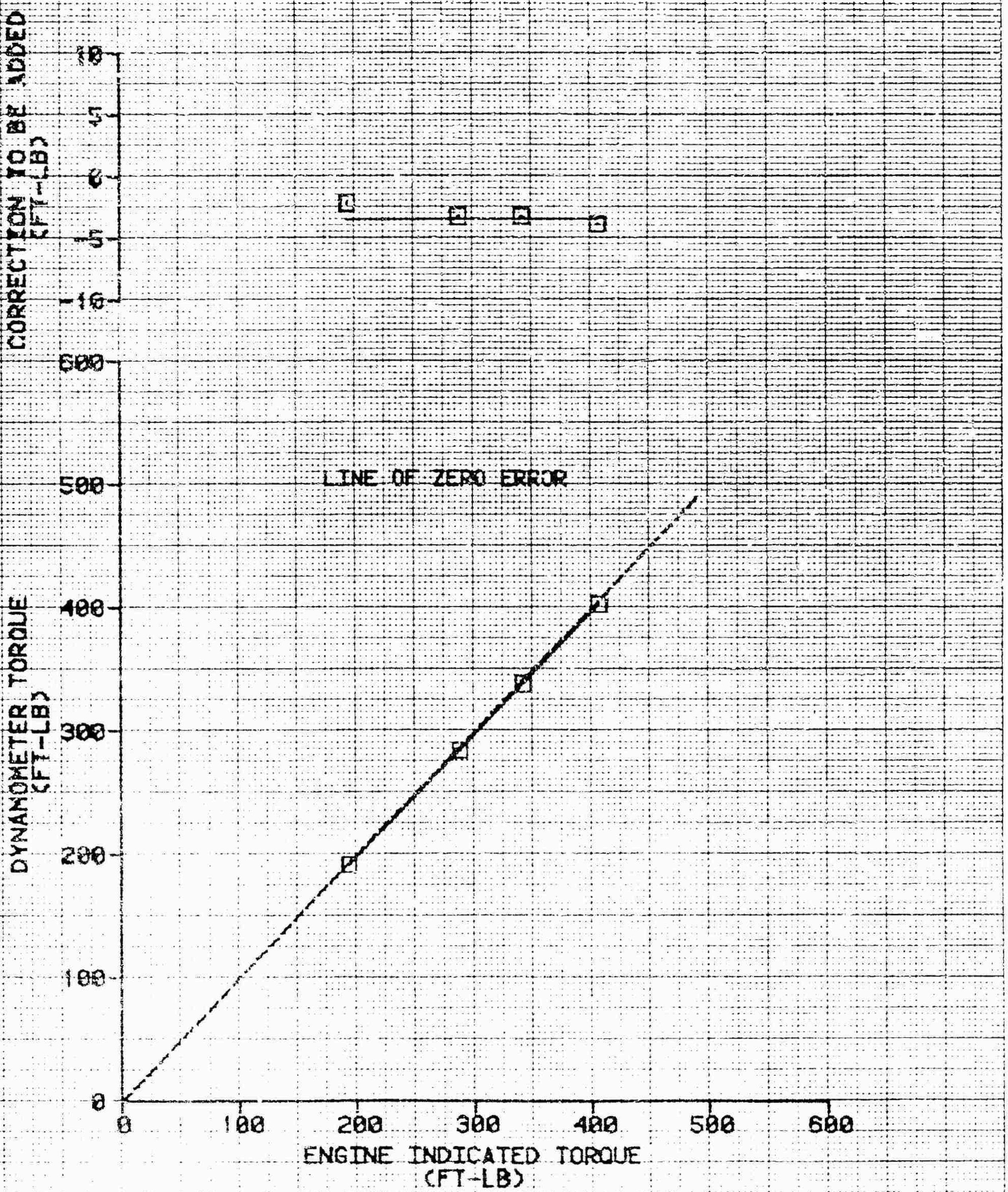


FIGURE 2
ENGINE TORQUEMETER CALIBRATION
T700-GE-701 S/N 374-114

NOTES: 1. POWER TURBINE SPEED = 20952 RPM
2. ENGINE INSTALLED ON RIGHT SIDE OF AIRCRAFT



coefficient and rotor speed. To maintain a constant ratio of gross weight to air density ratio (i.e., constant thrust coefficient) altitude was increased as fuel was consumed. Test day level flight power was corrected to standard day conditions by assuming that the test day dimensionless parameters, C_{P_t}, C_{T_t}, and μ_t were independent of atmospheric conditions. Consequently, the standard day dimensionless parameters C_{P_s}, C_{T_s}, and μ_s were indentical to C_{P_t}, C_{T_t}, and μ_t, respectively. From equation 1, the following relationship can be derived.

$$SHP_s = \frac{\rho_s}{\rho_t} \times SHP_t \qquad (6)$$

where:

t = Test day
s = Standard day

HANDLING QUALITIES

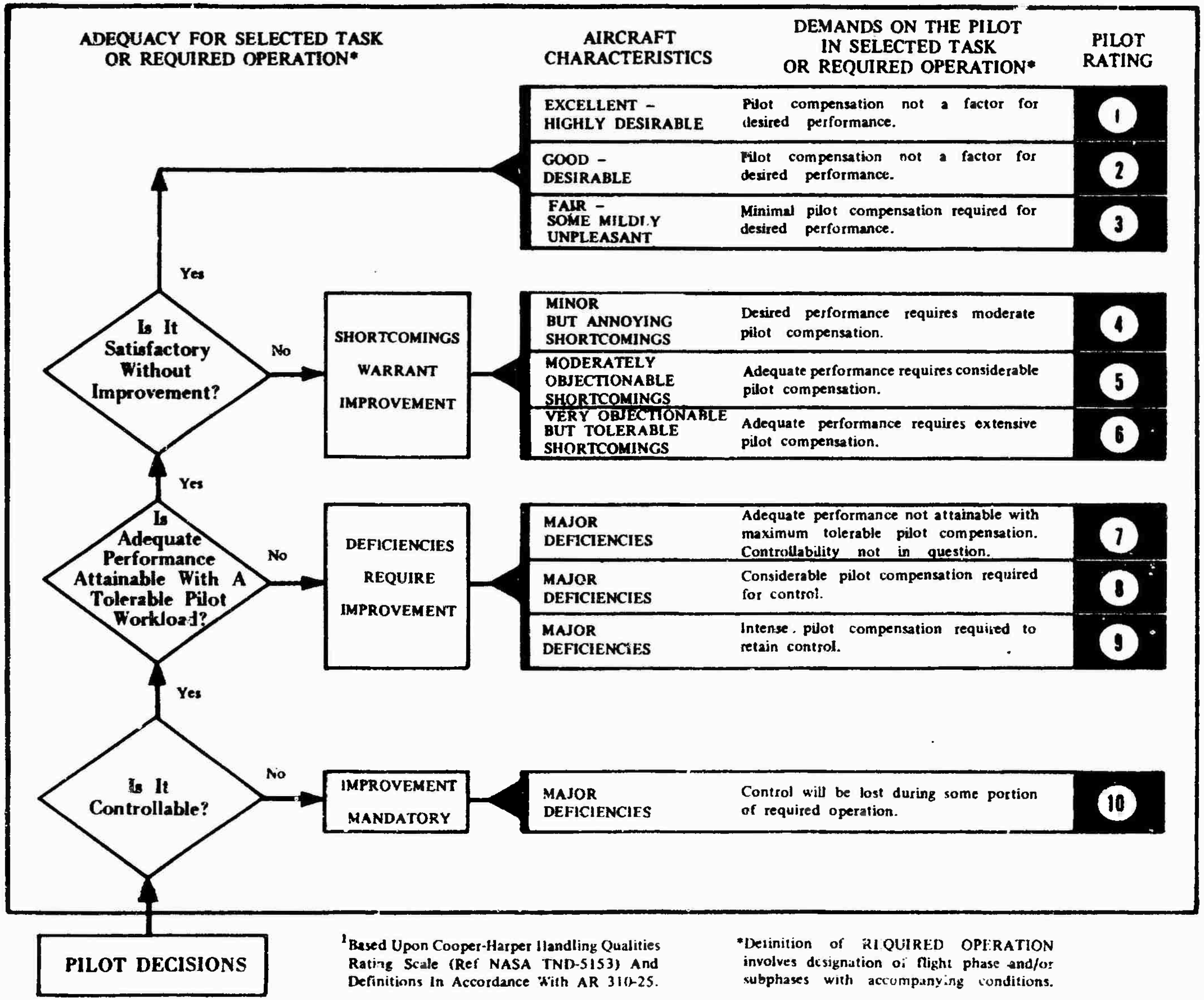
8. Stability and control data were collected and evaluated using standard test methods as described in reference 15, appendix A. The Handling Qualities Rating Scale presented in figure 3 was used to augment pilot comments relative to handling qualities. Turbulence reporting criteria used during this evaluation are presented in table 1.

VIBRATION

9. The vibration data were reduced by means of a fast Fourier transform from the analog flight tape. Peak vibration amplitudes were extracted from this analysis at selected harmonics of the main rotor frequency. The Vibration Rating Scale, presented in figure 4, was used to augment crew comments on aircraft vibration levels.

AIRSPEED SYSTEM CALIBRATION

10. The boom airspeed system was calibrated by using a pace aircraft to determine the airspeed and altimeter position errors.



[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 3. Handling Qualities Rating Scale

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 4. Vibration Rating Scale

Calibrated airspeed was obtained by correcting indicated values for instrument error and position error (fig. 1, app C). Altitude position error was calculated from the airspeed position error by assuming that all errors in the pitot-static system were in the static side. The following equation was used:

$$\Delta P_p = 1.4\, P_{a_o}\left(\frac{V_{ic}}{a_o}\right)\left[1 + 0.2\left(\frac{V_{ic}}{a_o}\right)^2\right]^{2.5}\left(\frac{\Delta V_{pc}}{a_o}\right)$$

$$+ 0.7\, P_{a_o}\left[1 + 0.2\left(\frac{V_{ic}}{a_o}\right)^2\right]^{1.5}\left[1 + 1.2\left(\frac{V_{ic}}{a_o}\right)^2\right]\left(\frac{\Delta V_{pc}}{a_o}\right)^2 \qquad (3)$$

Where:

ΔP_p = Static position error

P_{a_o} = Atmospheric pressure at standard-day sea level

V_{ic} = Instrument corrected indicated airspeed

a_o = Speed of sound at standard-day sea level

ΔV_{pc} = Measured airspeed position error.

DEFINITIONS

20. The following definitions of deficiencies and shortcomings were used during this evaluation.

a. 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.

b. Shortcoming - An imperfection or malfunction occurring during 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.

Table 1. Turbulence Reporting Criteria

Intensity	Aircraft Reaction	Reaction Inside Aircraft	Reporting Term-Definition
Light	Turbulence that momentarily causes slight erratic changes in altitude and/or attitude (pitch, roll, yaw). Report as Light Turbulence;* or Turbulence that causes slight, rapid and somewhat rhythmic bumpiness without appreciable changes in altitude or attitude. Report as Light Chop	Occupants may feel a slight strain against seat belts or shoulder straps. Unsecured objects may be displaced slightly.	
Moderate	Turbulence that is similar to Light Turbulene but of greater intensity. Changes in altitude and/or attitude occur but the aircraft remains in positive control at all times. It usually causes variations in indicated airspeed. Report as Moderate Turbulence.* or Turbulence that is similar to Light Chop but of greater intensity. It causes rapid bumps or jolts without appreciable changes in aircraft altitude or attitude. Report as Moderate Chop.	Occupants feel definite strains against seat belts or shoulder straps. Unsecured objects are dislodged.	Occasional - Less than 1/3 of the time Intermittent - 1/3 to 2/3 Continuous - More than 2/3
Severe	Turbulence that causes large, abrupt changes in altitude and/or attitude. It usually causes large variations in indicated airspeed. Aircraft may be momentarily out of control. Report as Severe Turbulence.*	Occupants are forced violently against seat belts or shoulder straps. Unsecured objects are tossed about.	
Extreme	Turbulence in which the aircraft is violently tossed about and is practically impossible to control. It may cause structural damage. Report as Extreme Turbulence.*		

APPENDIX E. TEST DATA

INDEX

FIGURE 1

NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

NOTES:
1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 208.4 (AFT)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 4 THROUGH 8

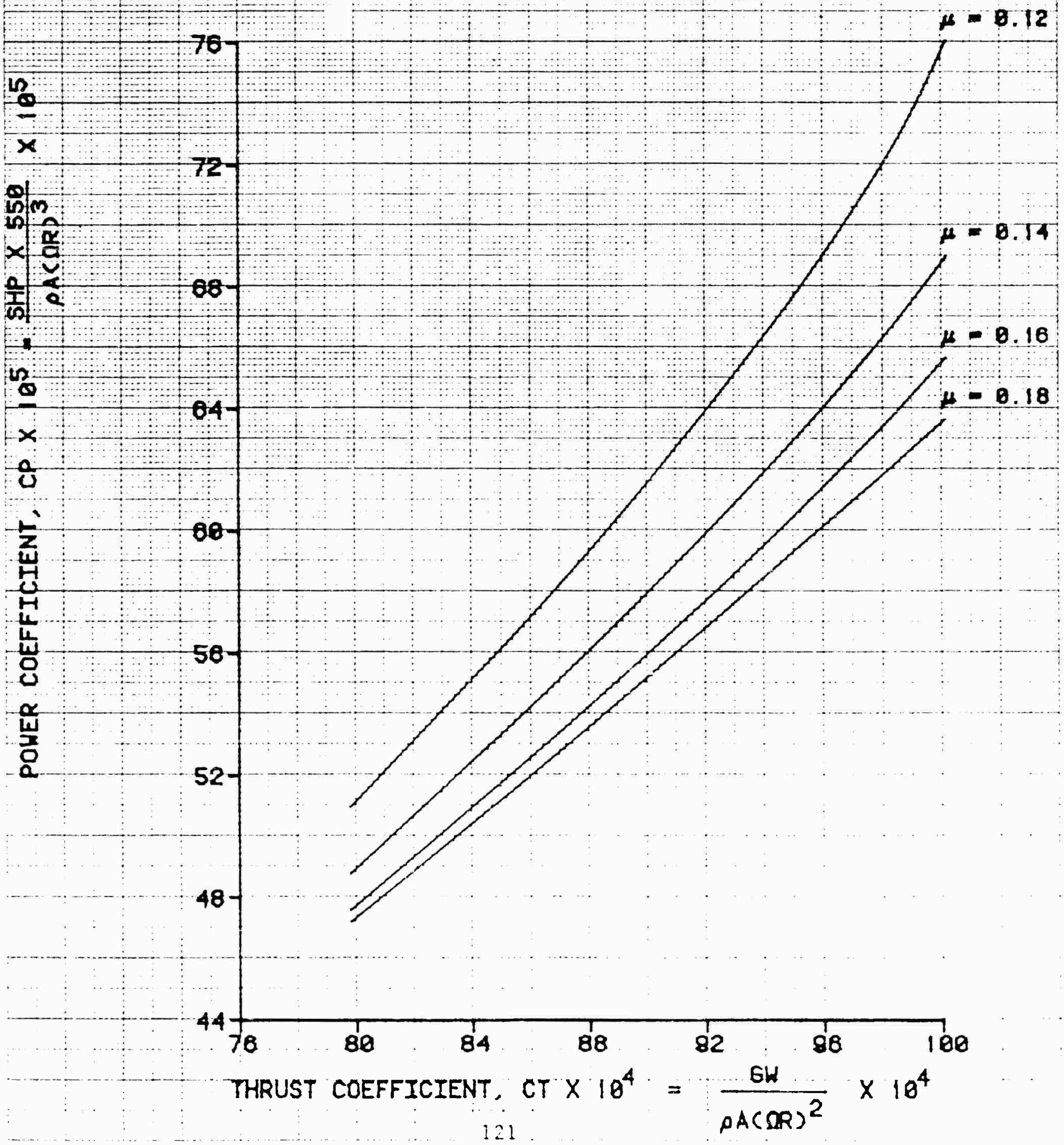
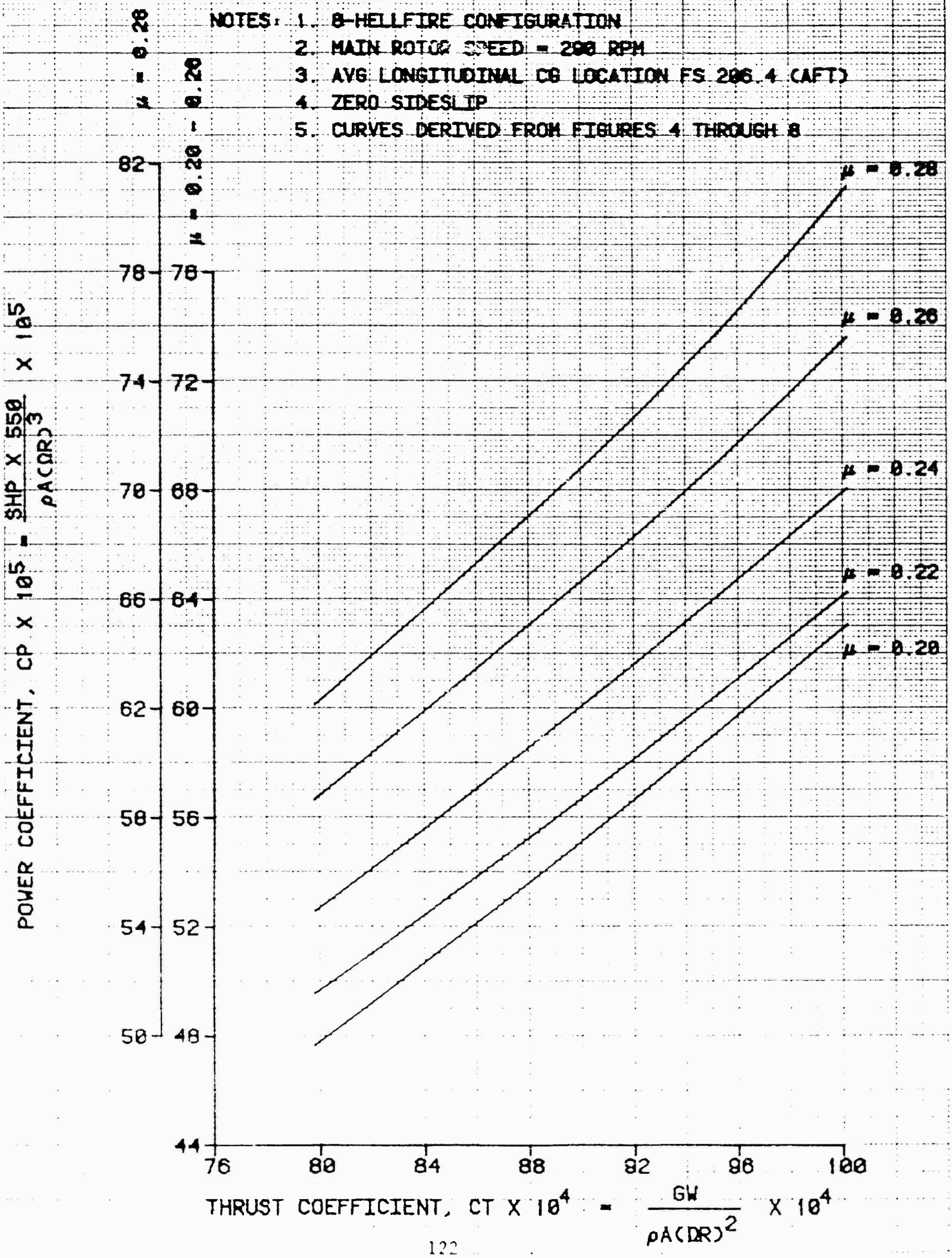


FIGURE 2
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355



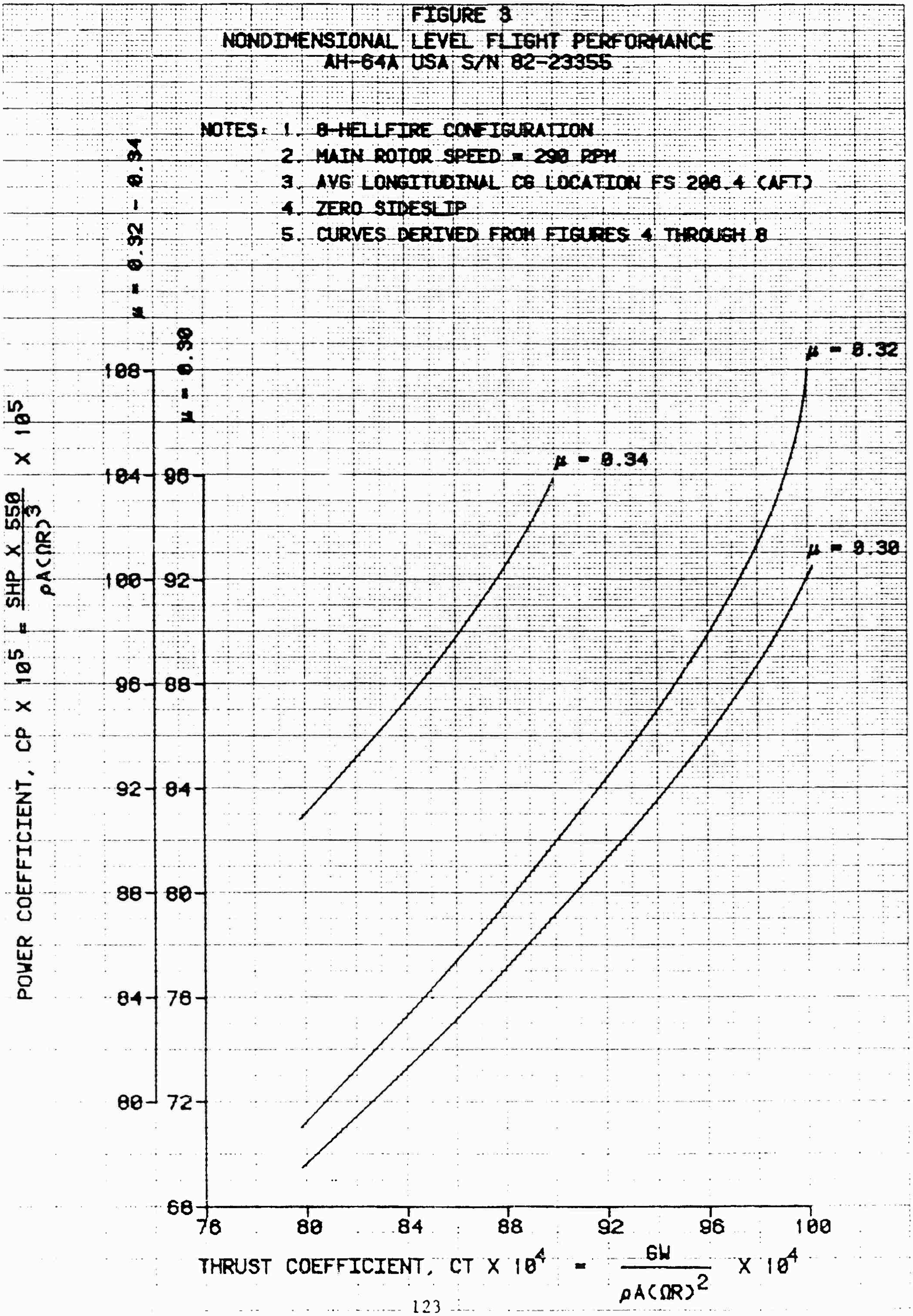


FIGURE 8
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355
NOTES: 1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CG LOCATION FS 208.4 (AFT)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 4 THROUGH 8
μ = 0.32 - 0.34
μ = 0.30
μ = 0.32
μ = 0.34
μ = 0.30
POWER COEFFICIENT, CP X 10^5 = SHP X 550 / ρA(ΩR)^3 X 10^5
108
104
100
96
92
88
84
80
96
92
88
84
80
76
72
68
76
80
84
88
92
96
100
THRUST COEFFICIENT, CT X 10^4 = GW / ρA(ΩR)^2 X 10^4

FIGURE 4
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
14530	206.2 (AFT)	7530	18.5	288	0.007991	8-HELLFIRE

NOTE: ZERO SIDESLIP

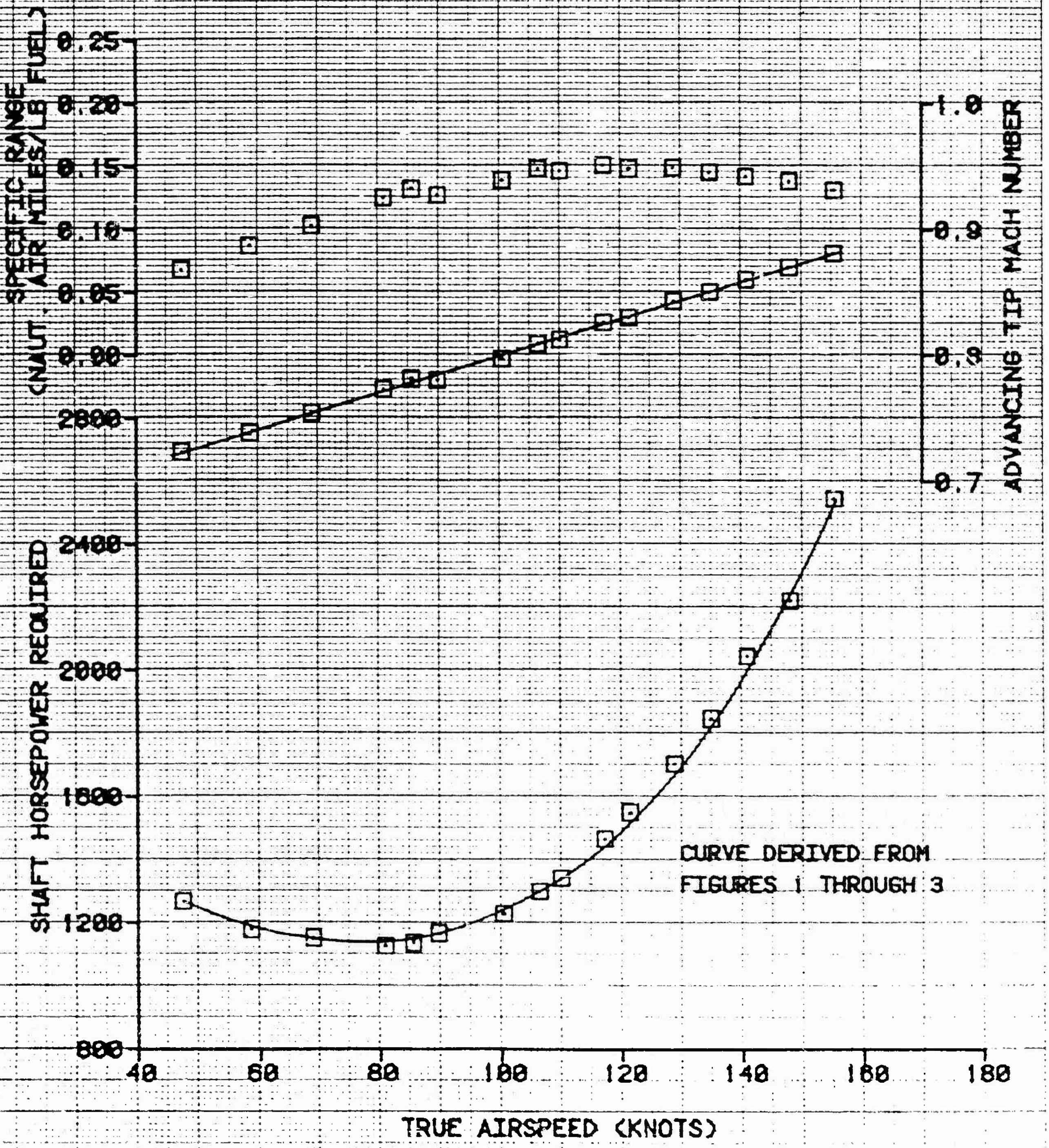


FIGURE 5
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
14400	200.6 (AFT)	9620	18.0	290	0.000480	8-HELLFIRE

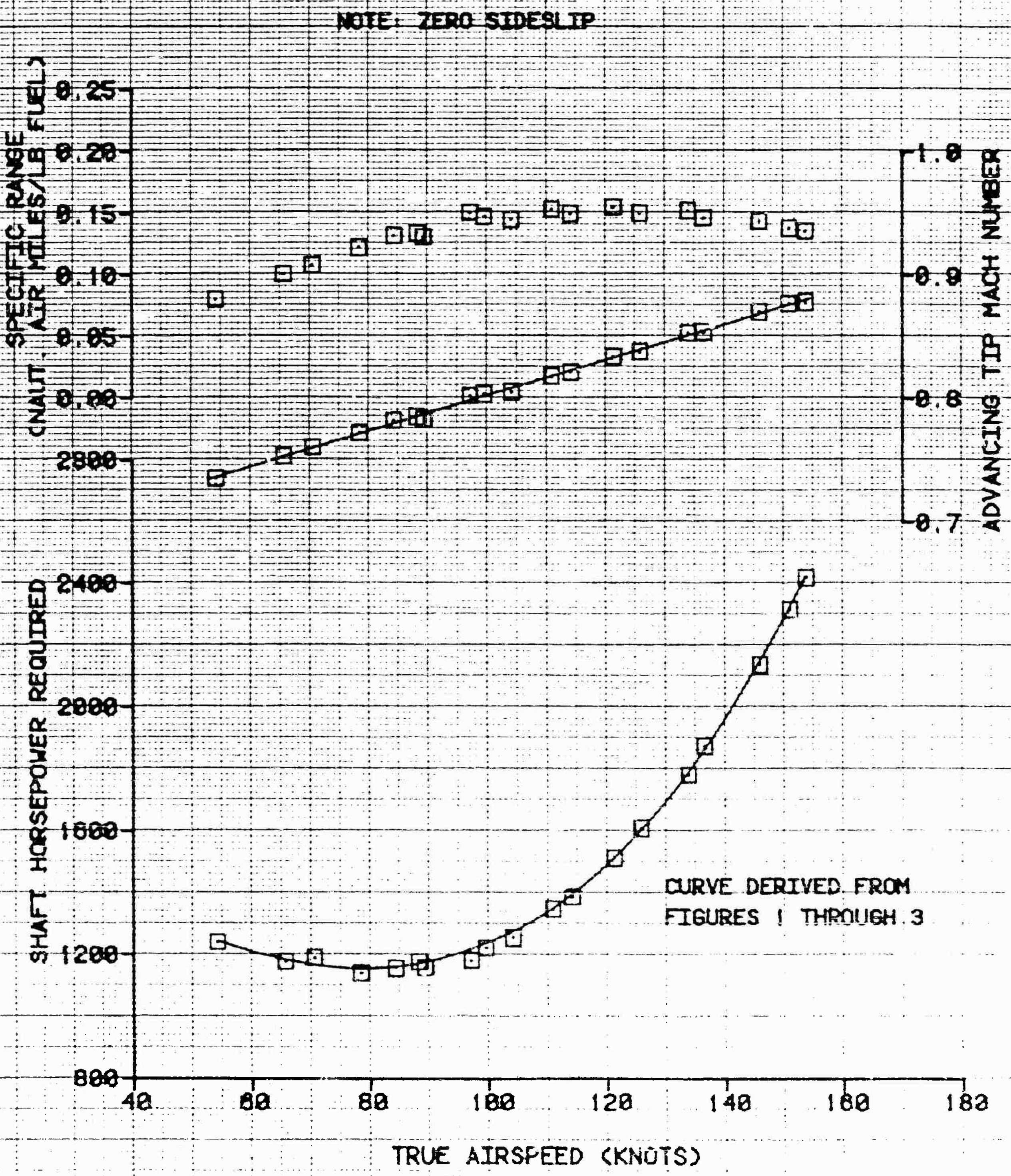


FIGURE 8

LEVEL FLIGHT PERFORMANCE

AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
14770	200.7 (AFT)	10000	13.5	290	0.000978	8-HELLFIRE

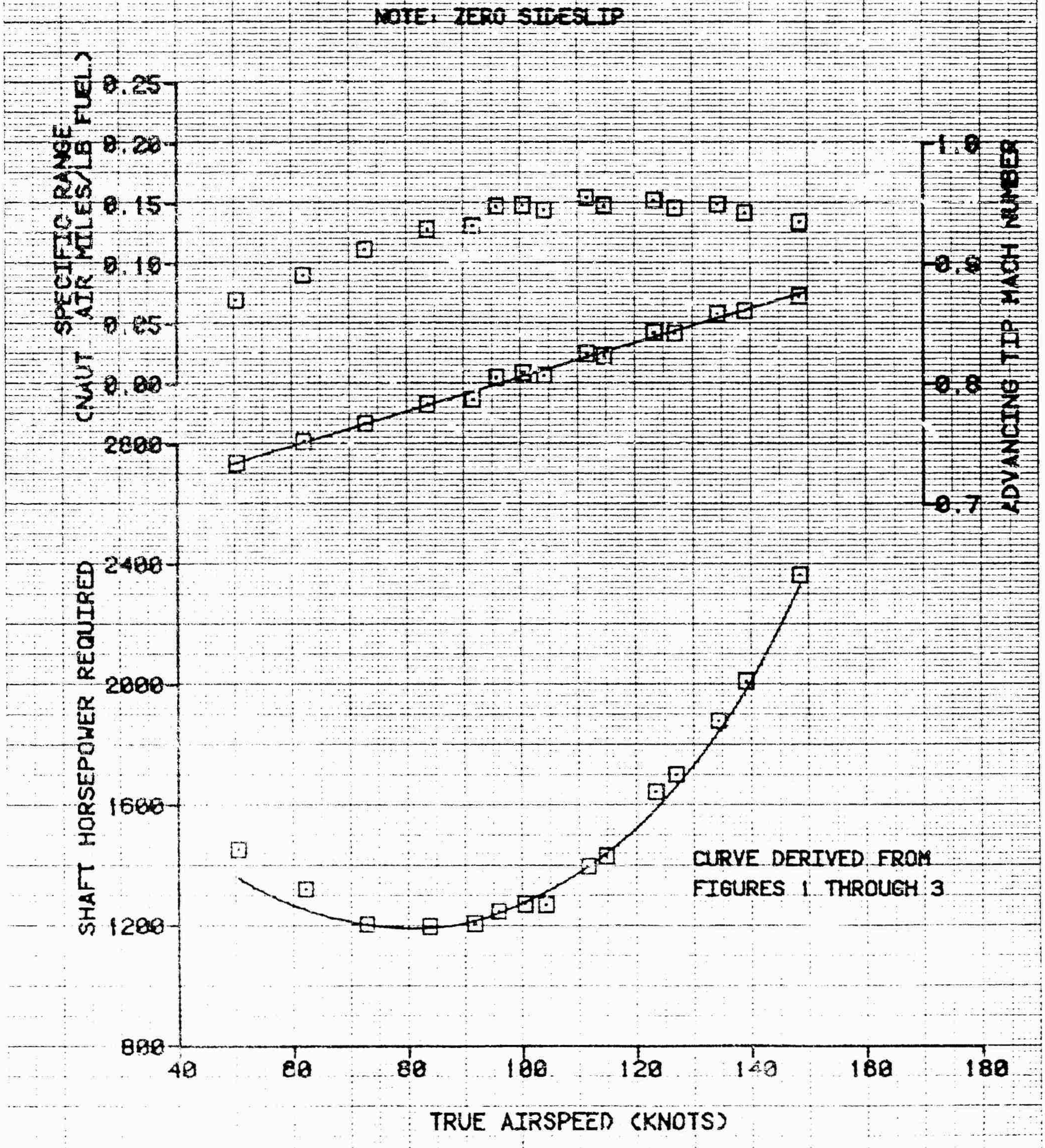


FIGURE 7
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
14850	208.2 (AFT)	12020	0.5	290	0.008525	8-HELLFIRE

NOTE: ZERO SIDESLIP

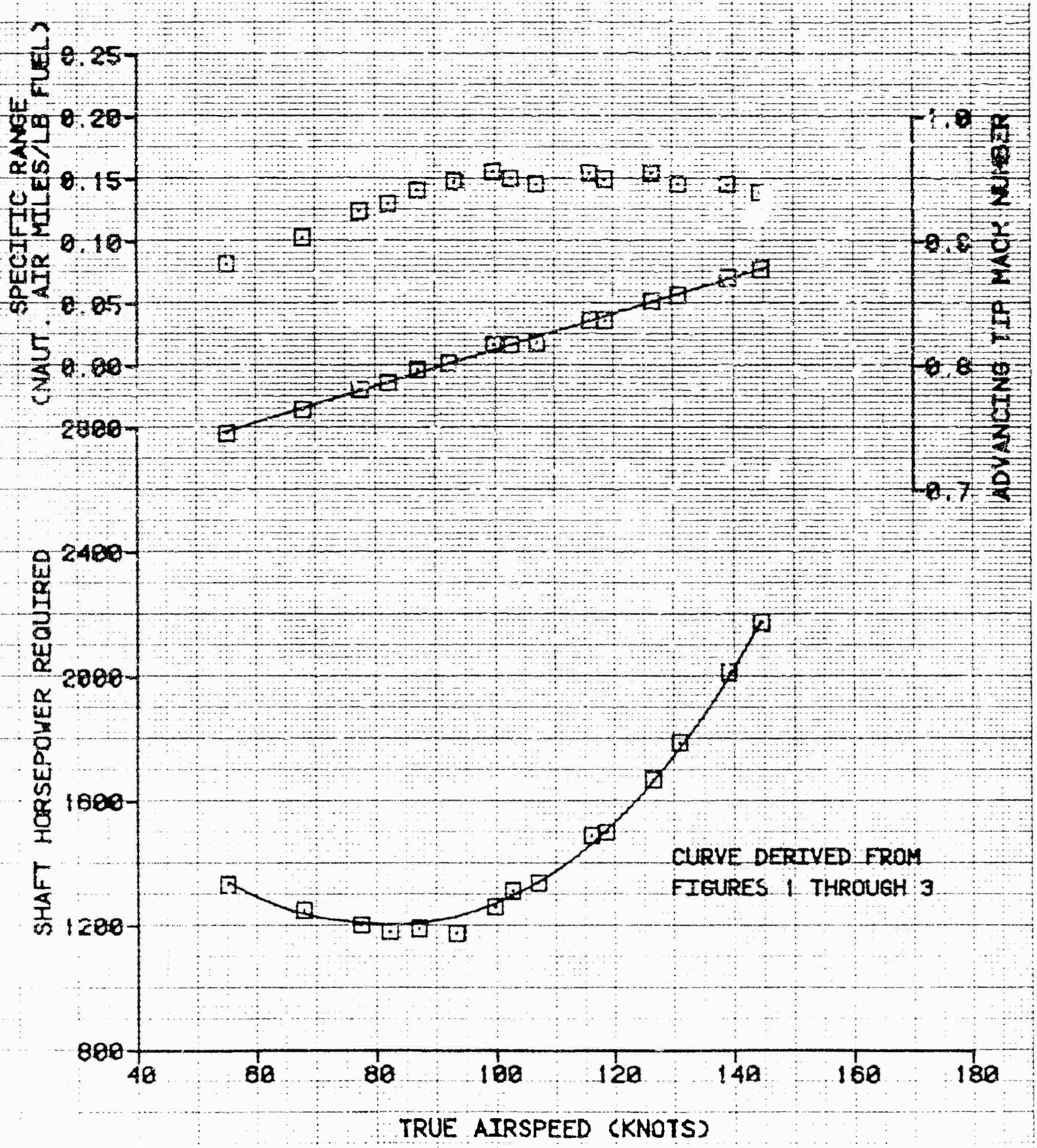


FIGURE 8
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIGURATION
15020	200.1 (AFT)	13680	8.0	290	0.010002	8-HELLFIRE

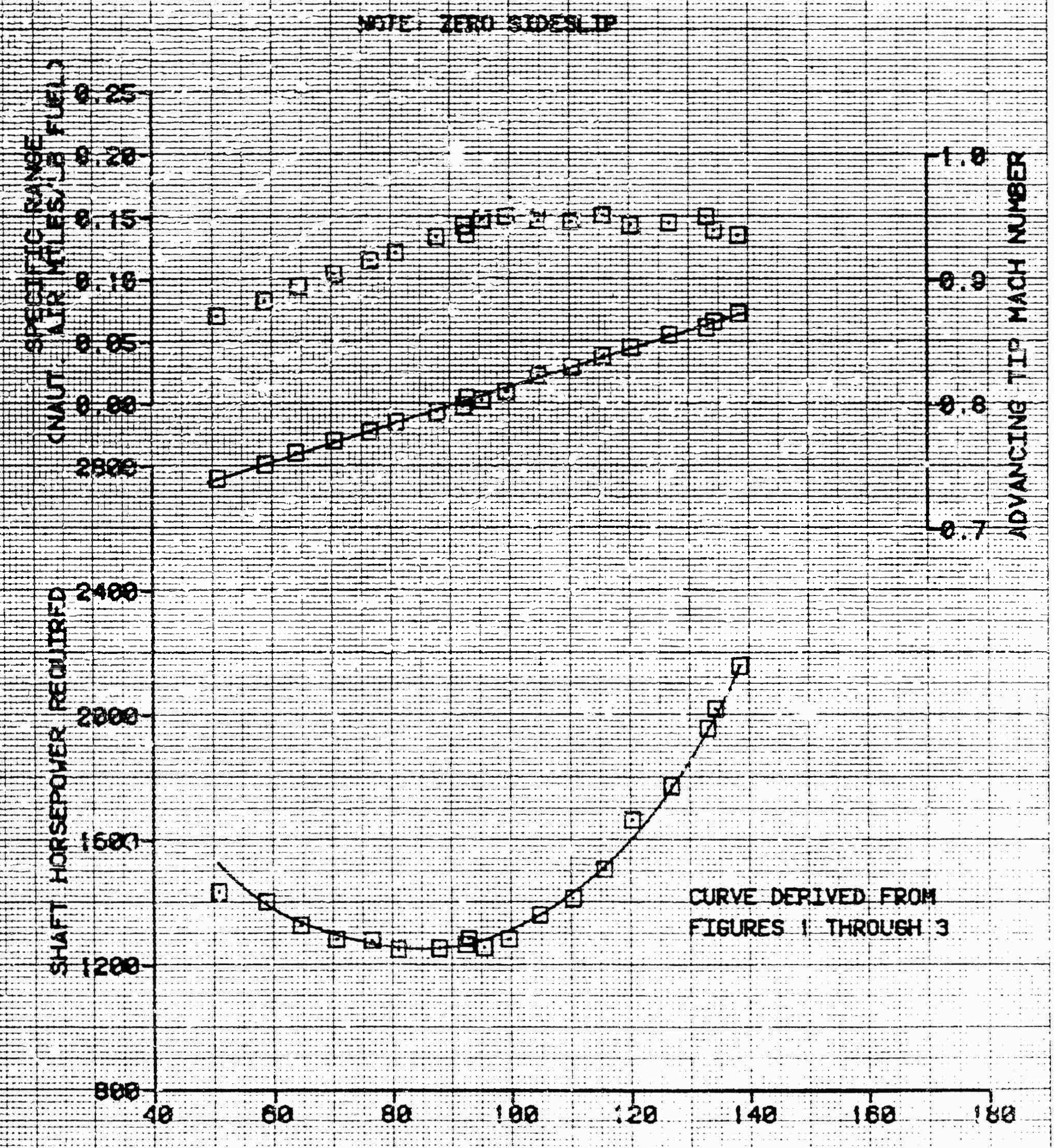


FIGURE 9
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFF.	CONFIG.
14694	206.4 (AFT)	7120	35.0	289	0.000018	8-HELLFIRE

NOTE: DASHED LINE DENOTES A&FC S DATA CORRECTED FOR A DIFFERENCE IN LONGITUDINAL CG (-3.1 SQUARE FEET, EQUIVALENT FLAT PLATE AREA)

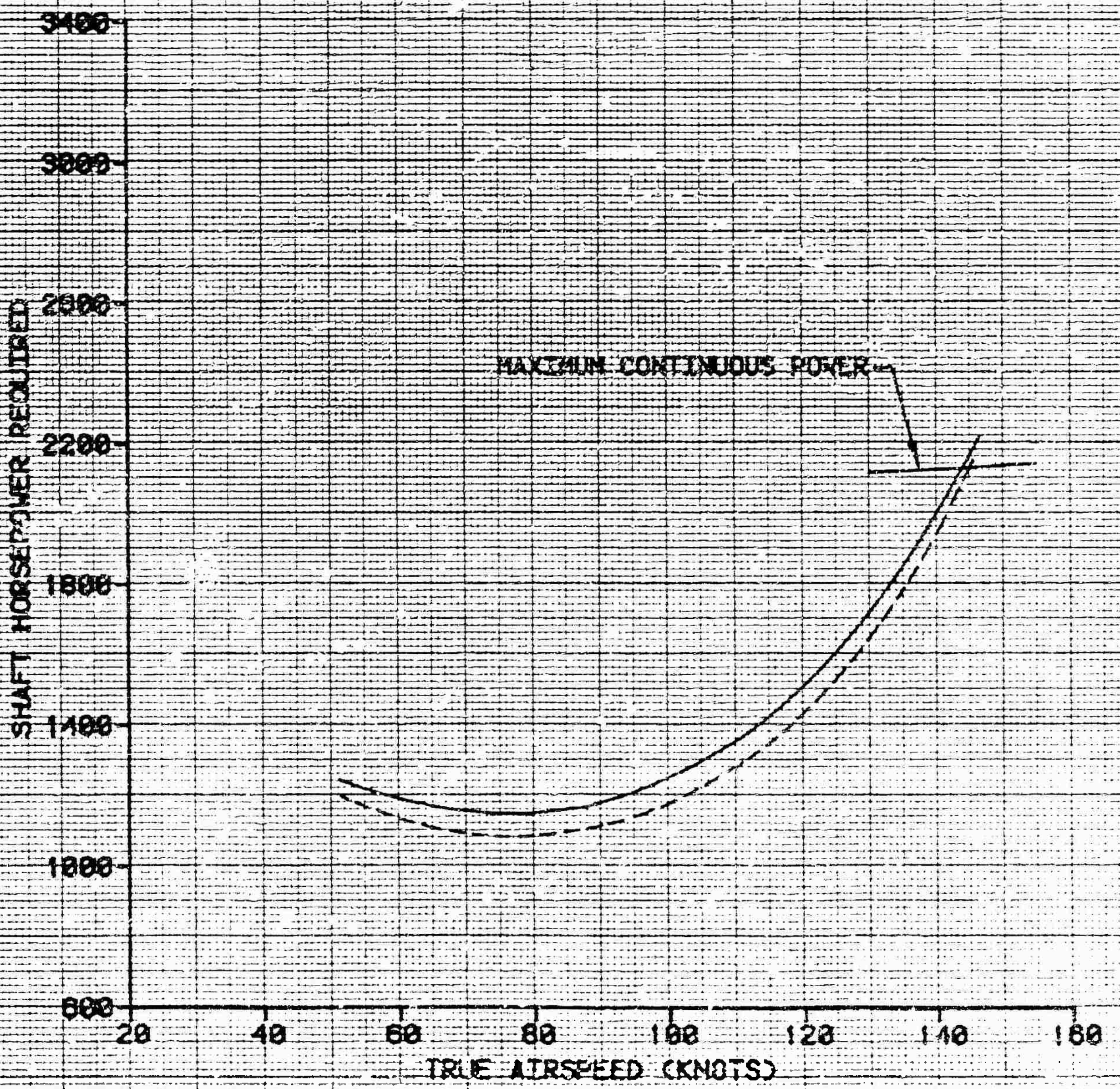


FIGURE 10

LIMITS OF CYCLIC CONTROL TRAVEL
AH-64A USA S/N 82-23355

NOTES: 1. ROTORS STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS
3. CONTROL POSITIONS MEASURED AT CENTER OF GRIP

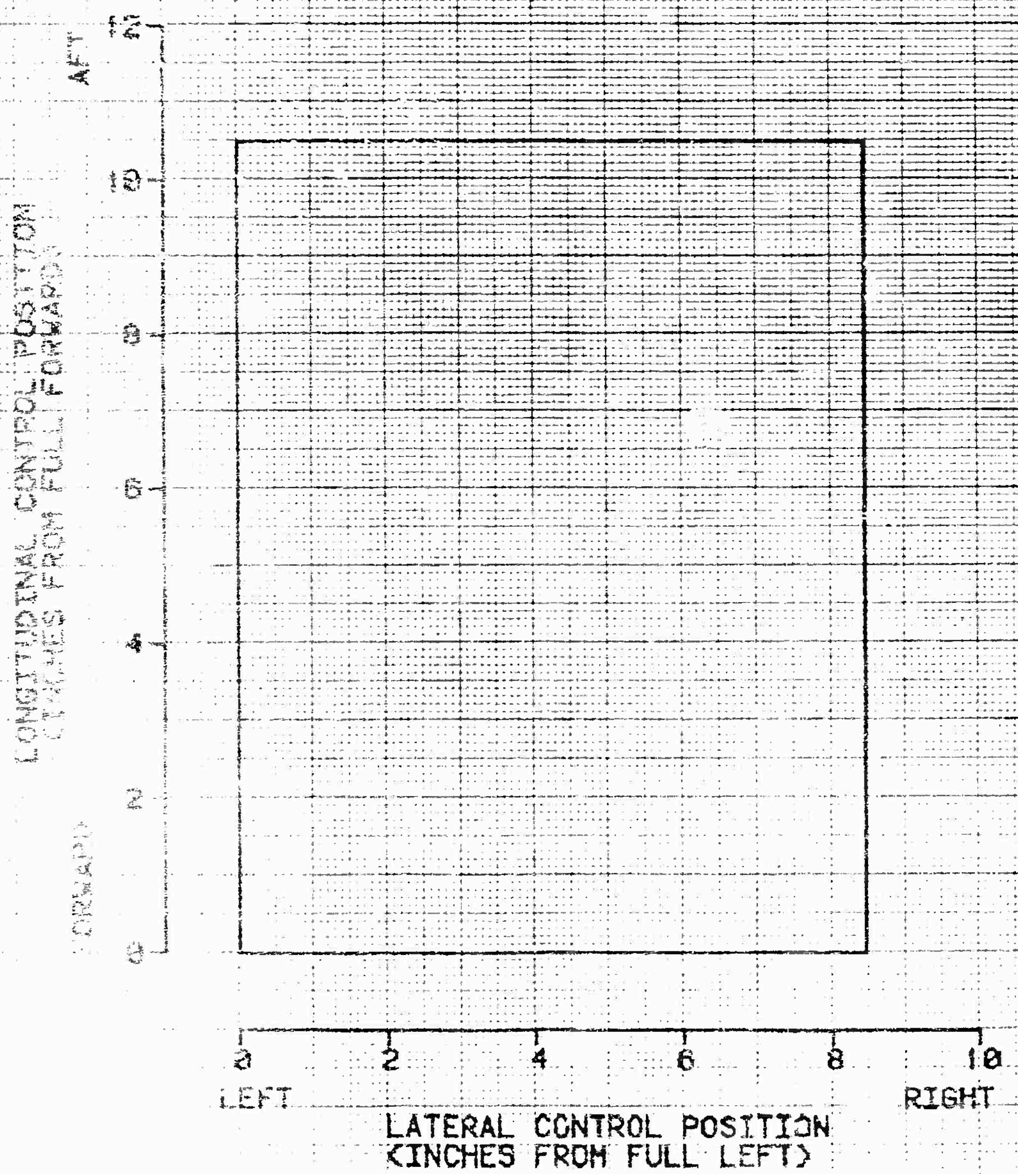


FIGURE 11
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
AH-64A USA S/N 82-23355

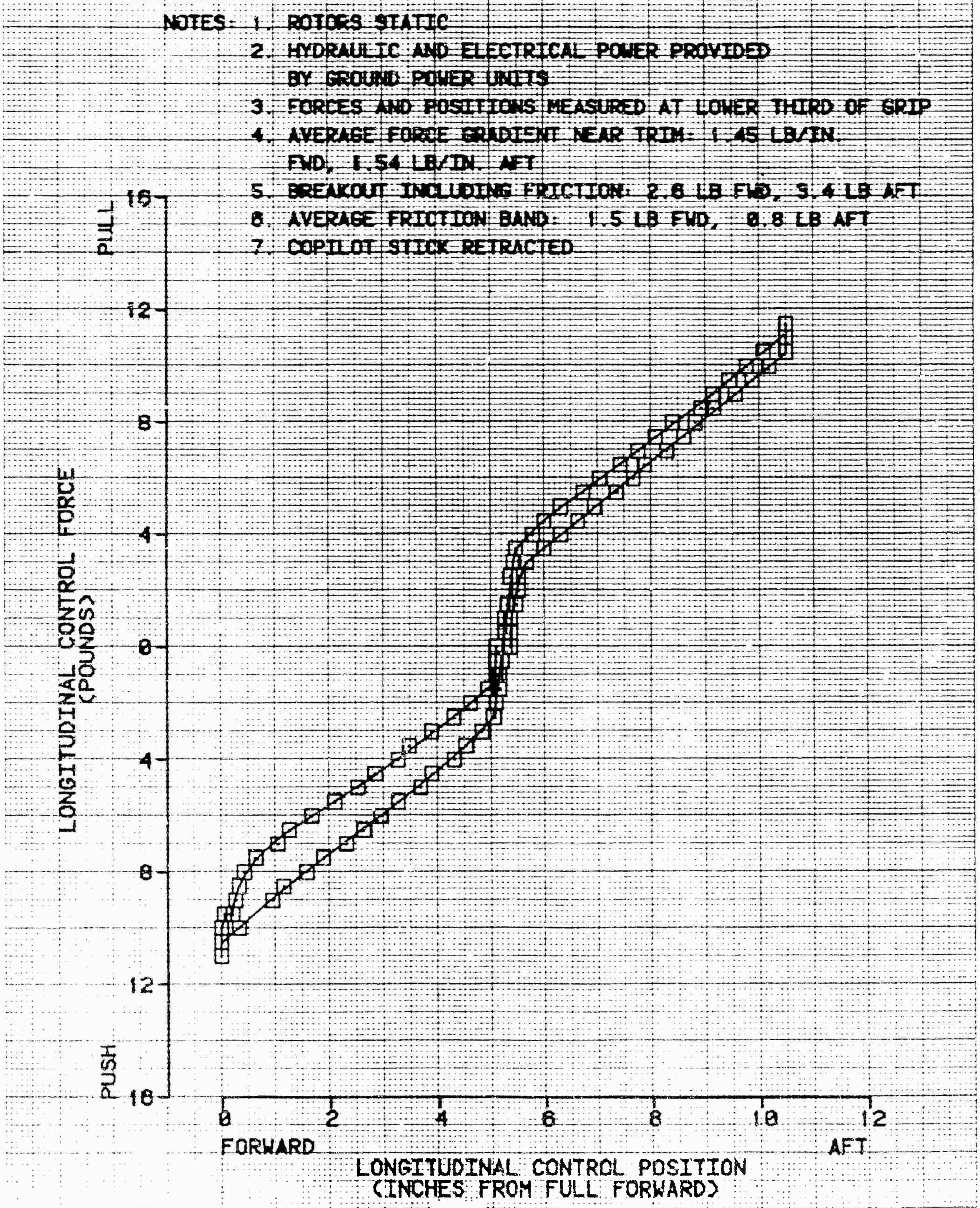


FIGURE 12
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
AH-64A USA S/N 82-23355

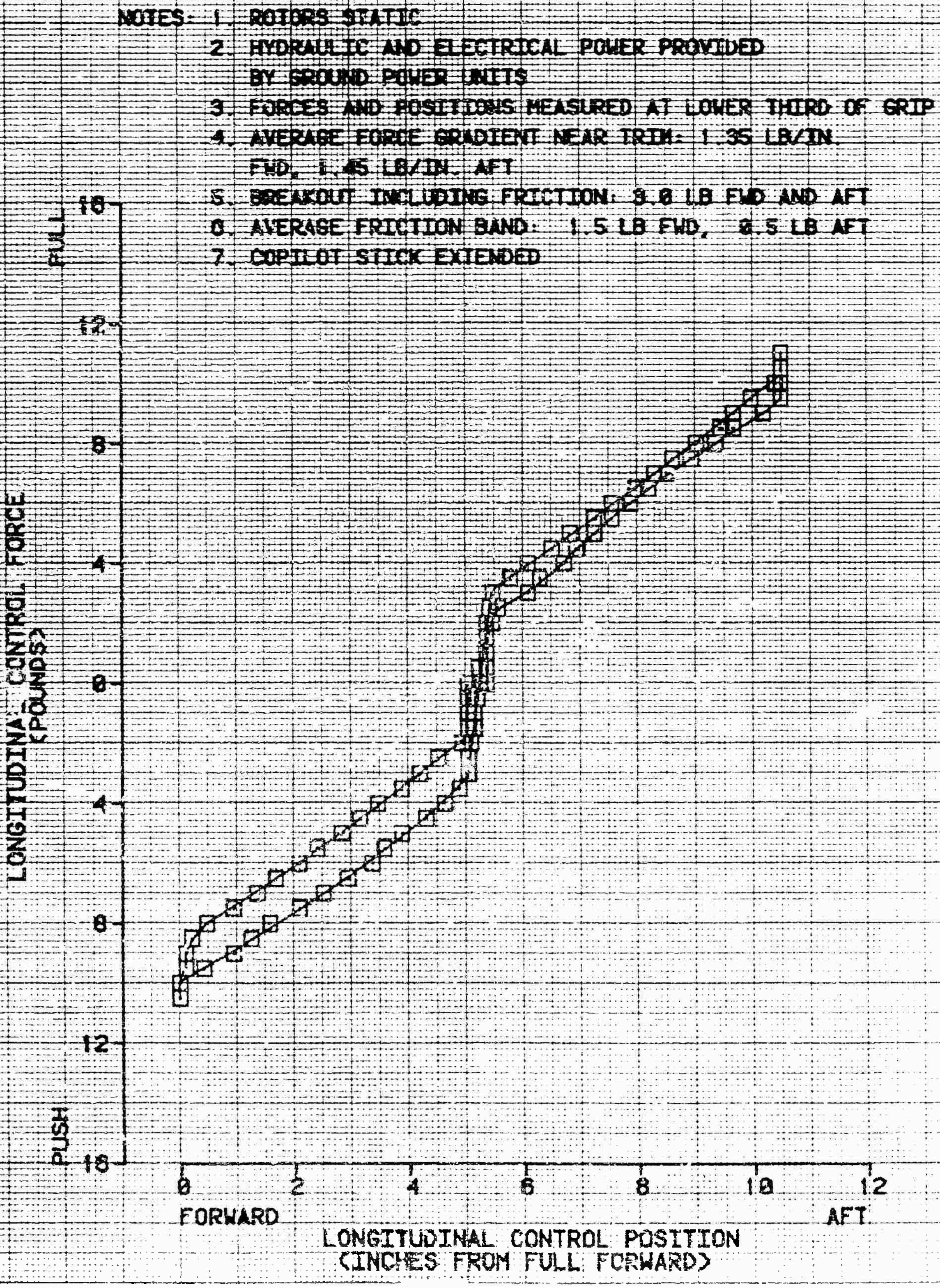


FIGURE 13
LATERAL CONTROL SYSTEM CHARACTERISTICS
AH-64A USA S/N 82-23355

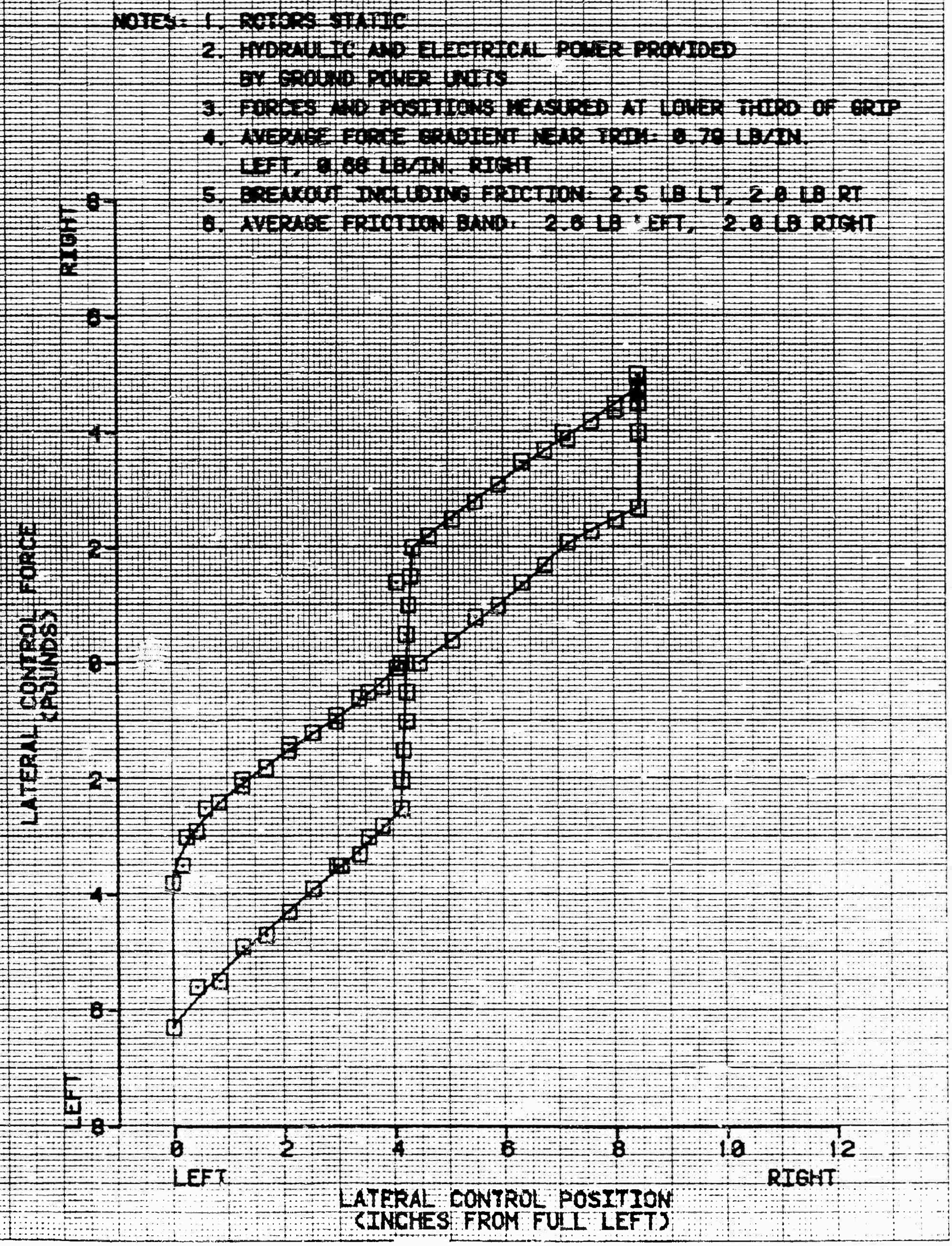


FIGURE 14
DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS
AH-64A USA S/N 82-23355

NOTES: 1. ROTORS STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS
3. FORCES AND POSITIONS MEASURED AT CENTER OF PEDAL
4. AVERAGE FORCE GRADIENT NEAR TRIM: 1.7 LB/IN. LEFT, 2.0 LB/IN. RIGHT
5. BREAKOUT INCLUDING FRICTION: 9.0 LB LT, 6.0 LB RT
6. AVERAGE FRICTION BAND: 10.5 LB LEFT, 10.2 LB RIGHT

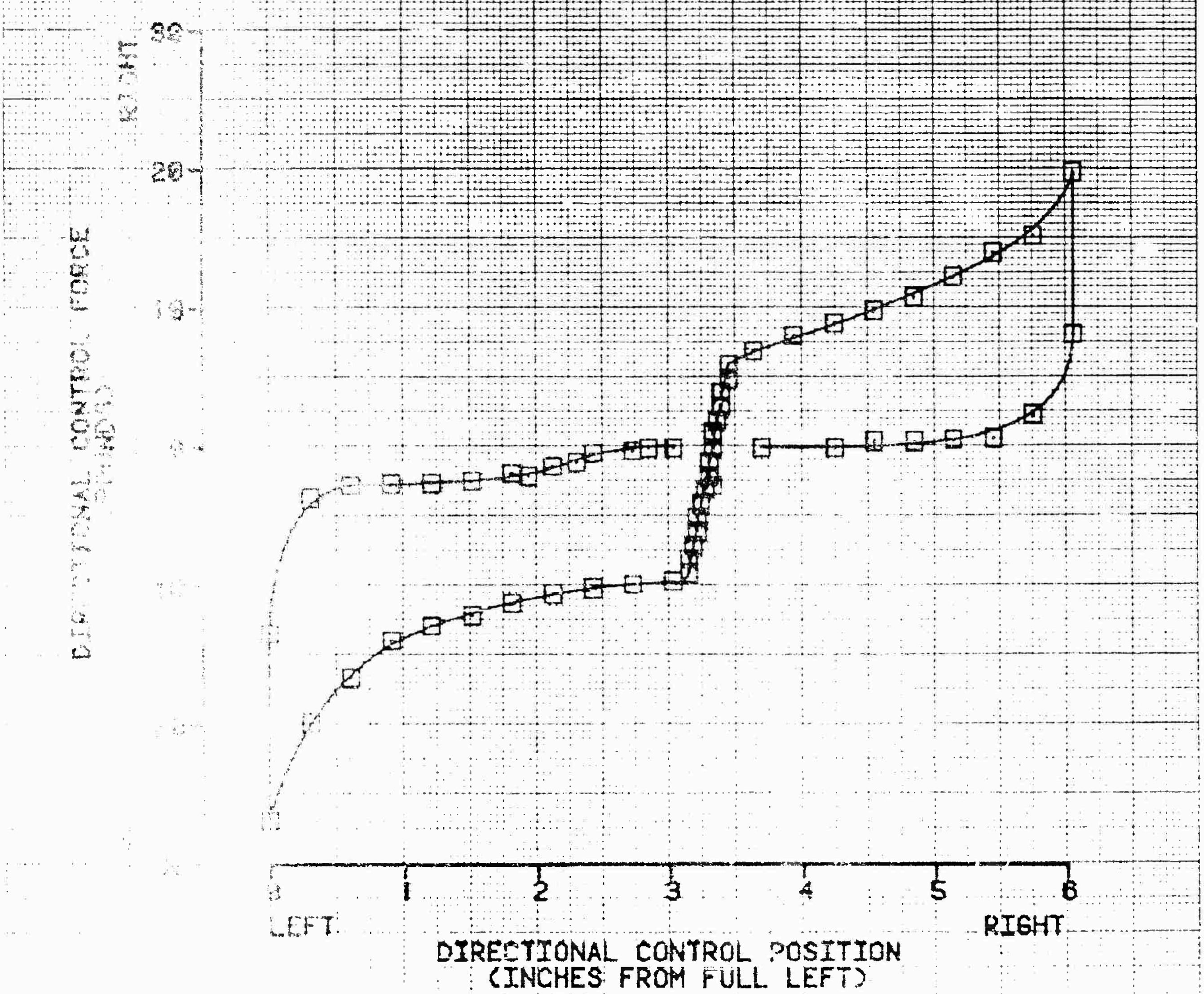


FIGURE 15
CONTROL SYSTEM IMBALANCE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14770	206.4 (AFT)	7280	25.0	290	LEVEL

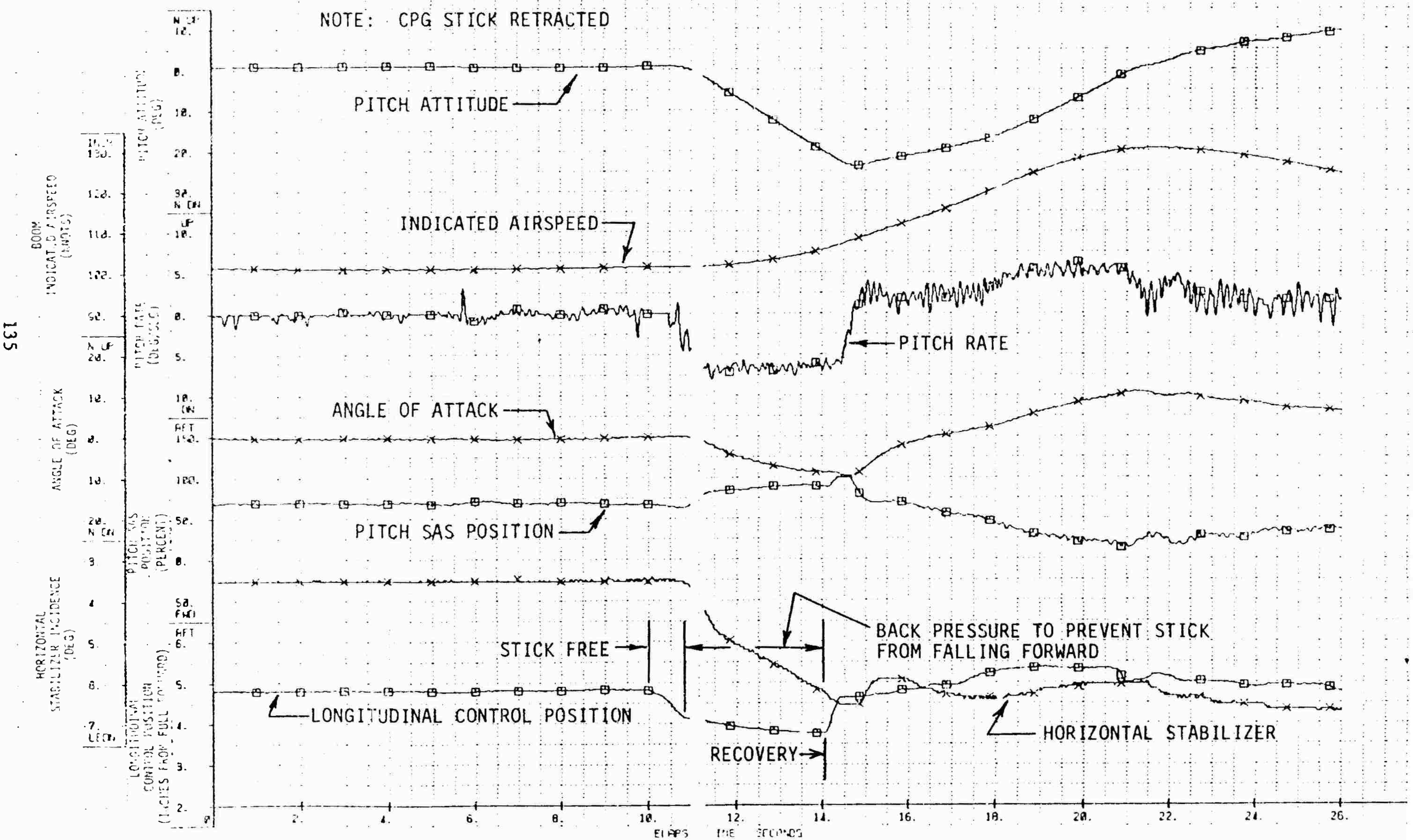


FIGURE 16
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
AH-64A USA S/N 82-23355

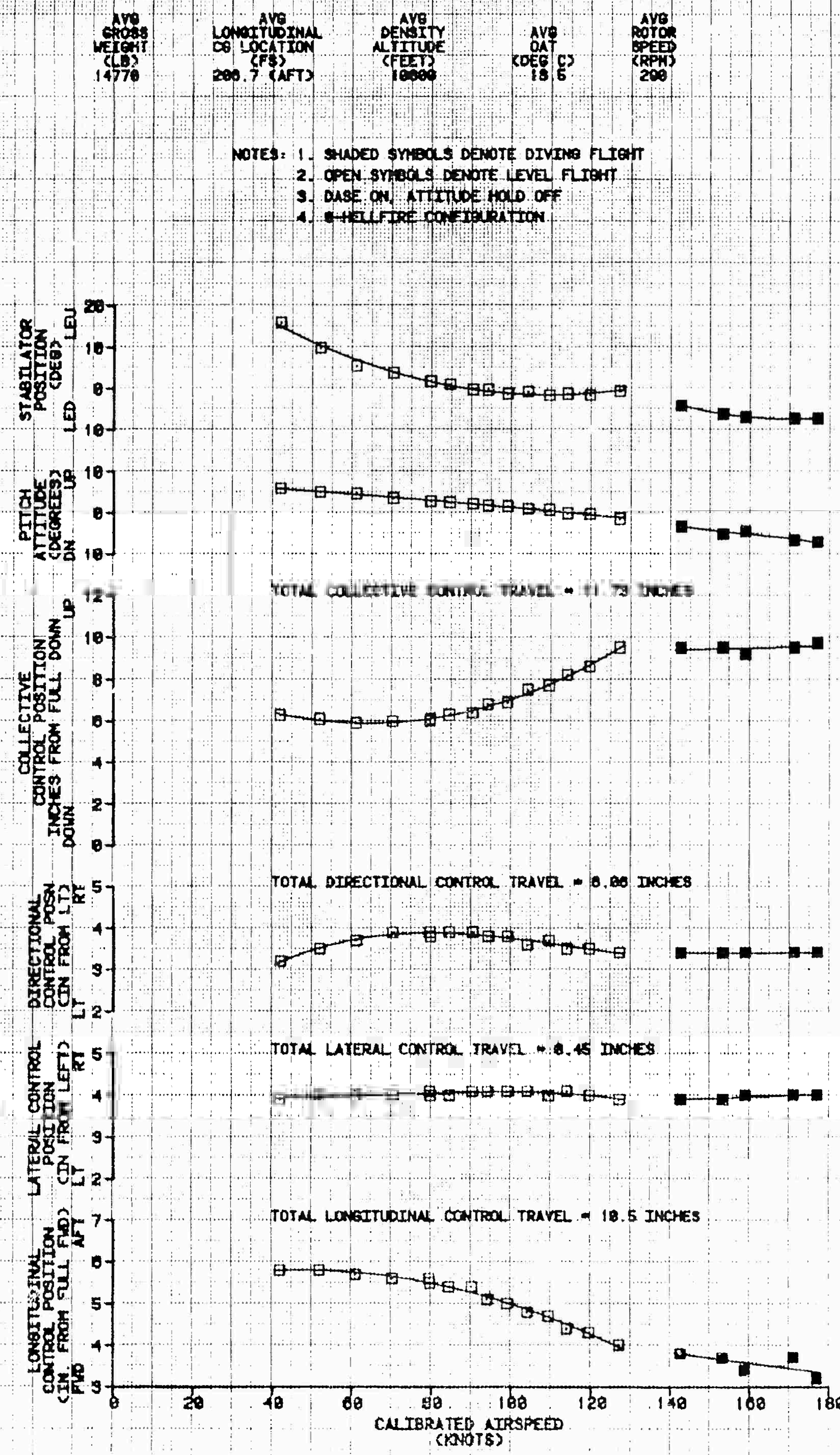


FIGURE 17
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
AH-64A USA S/N 82-23355

SYMBOL	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
□	14790	200.5 (AFT)	7430	24.0	289	CLIMB
△	14790	200.5 (AFT)	7390	24.0	293	DESCENT

NOTES: 1. DASE ON, ATTITUDE HOLD OFF
2. 8-HELLFIRE CONFIGURATION

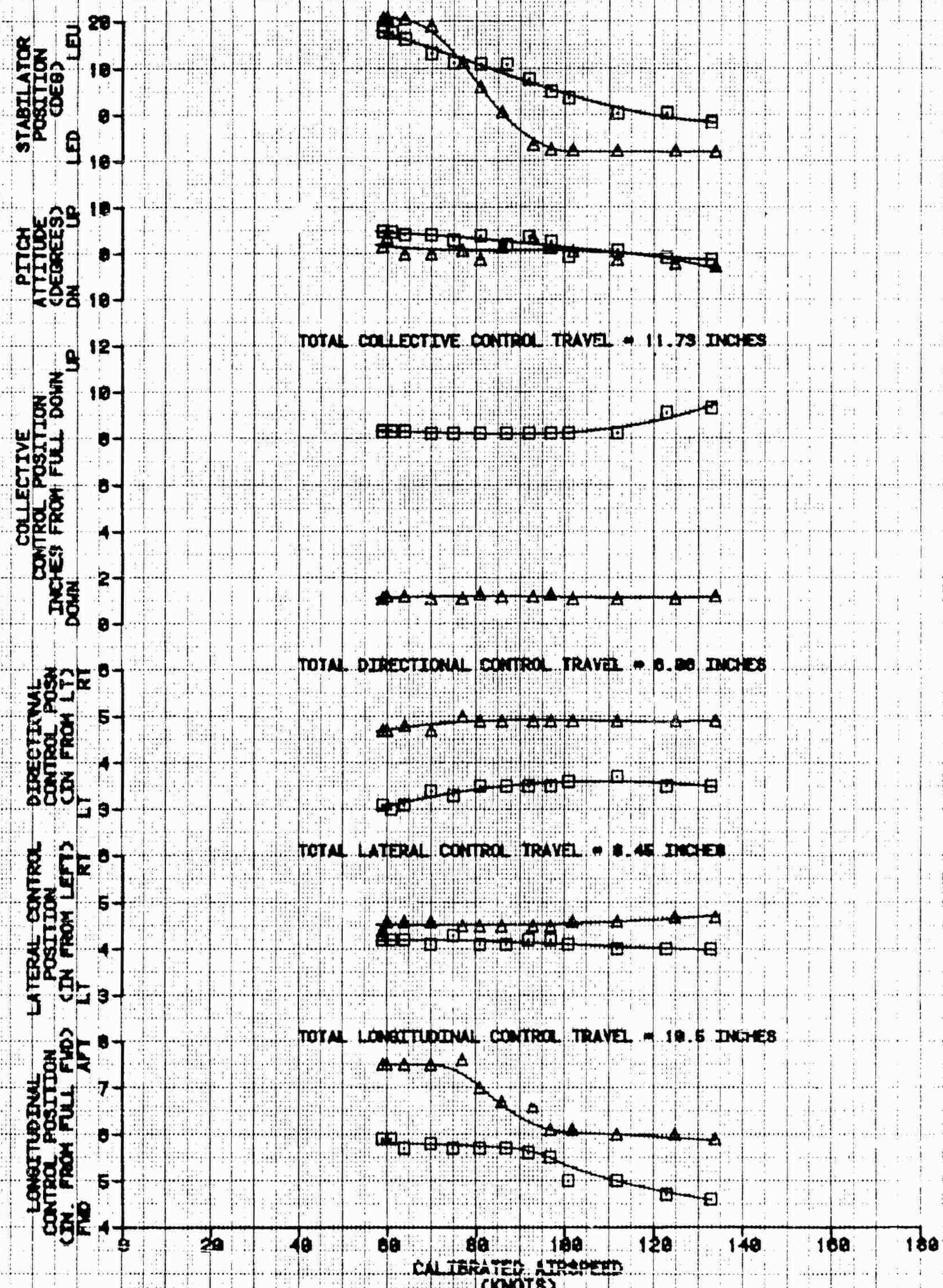
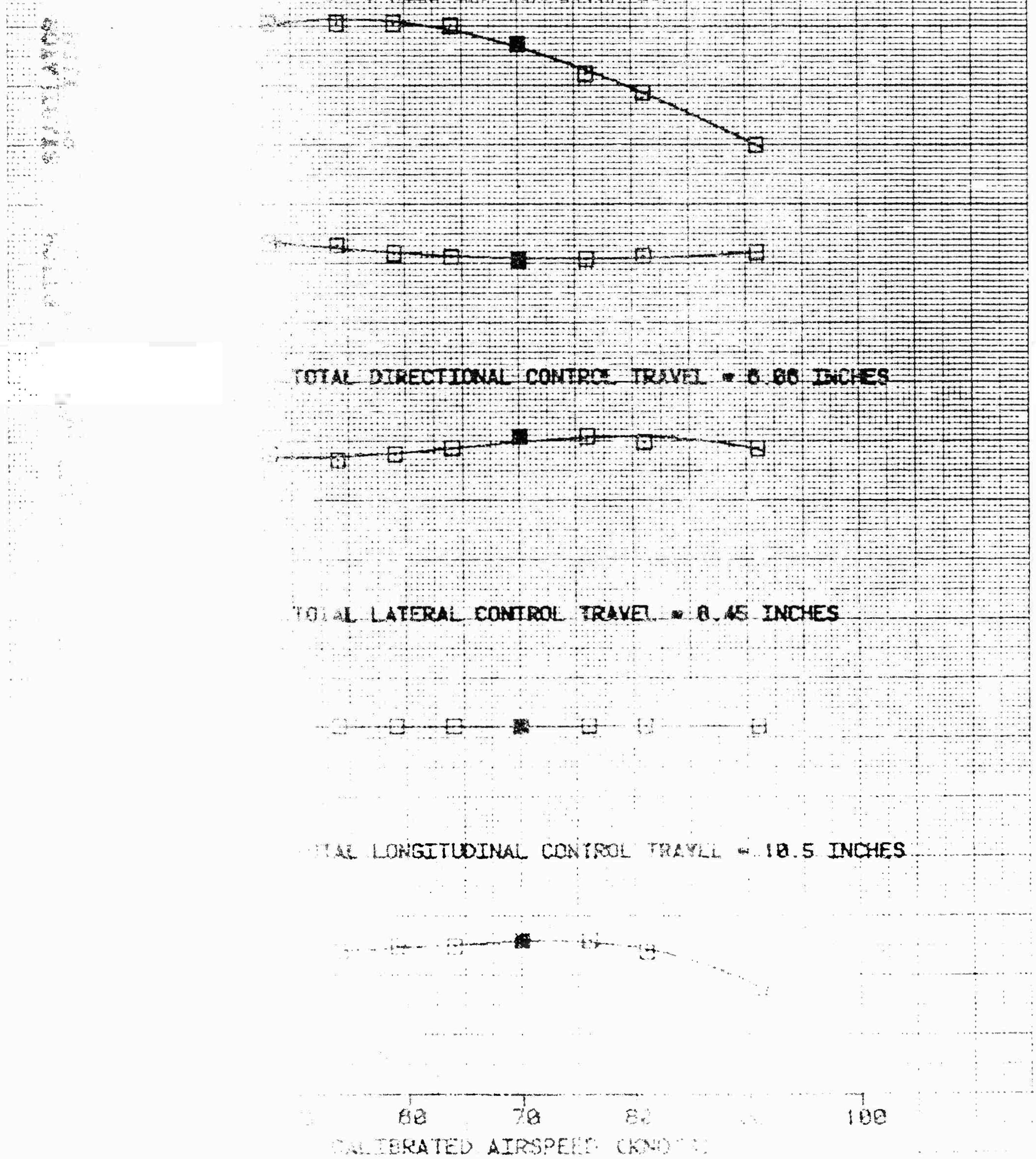


FIGURE 18
STATIC LONGITUDINAL STABILITY
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
[illegible]	206.5 (AFT)	7140	24.0	290	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. DASE ON, ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION



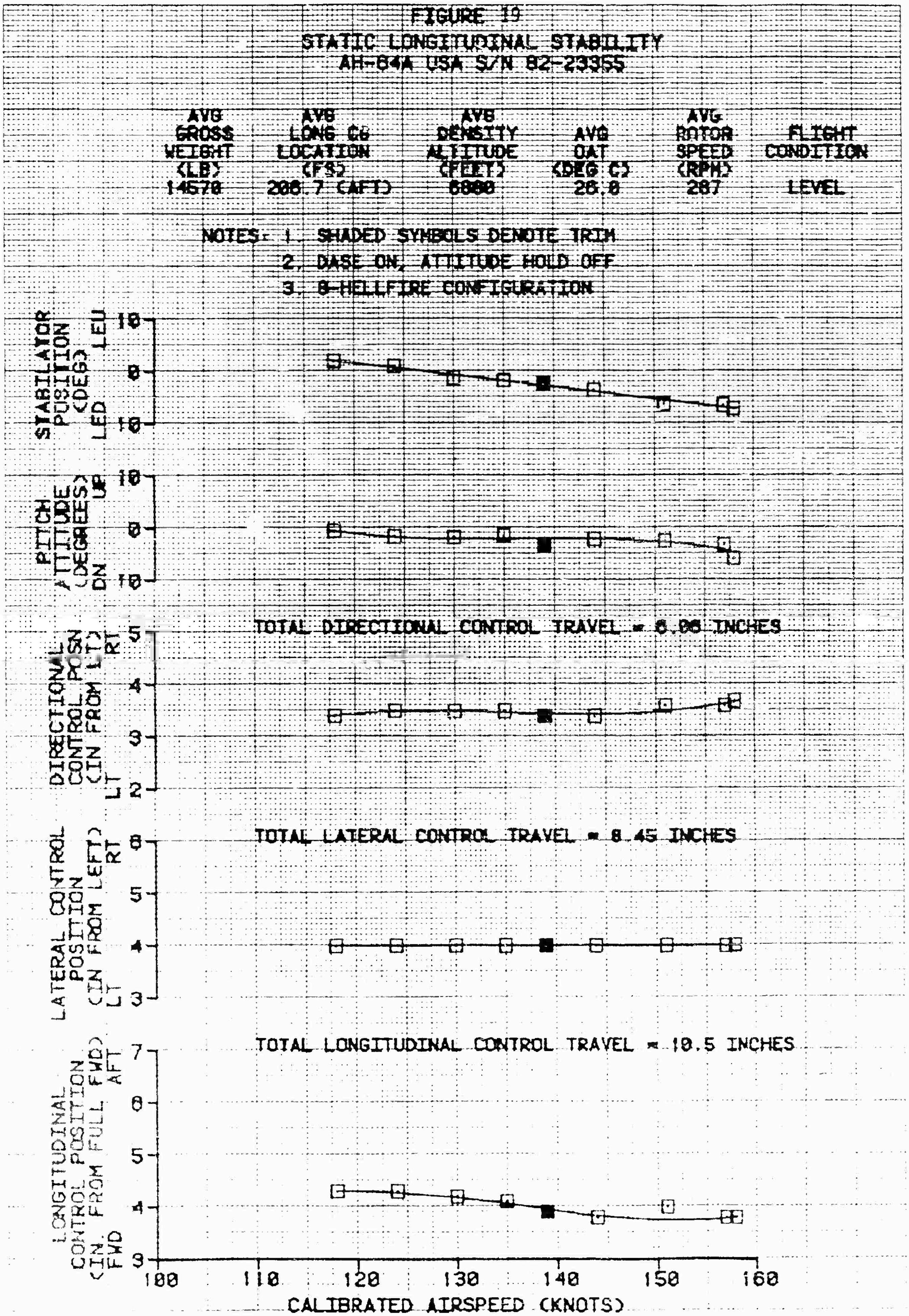


FIGURE 19
STATIC LONGITUDINAL STABILITY
AH-64A USA S/N 82-23355
AVG GROSS WEIGHT (LB) 14570
AVG LONG CG LOCATION (FS) 208.7 (AFT)
AVG DENSITY ALTITUDE (FEET) 6800
AVG OAT (DEG C) 26.0
AVG ROTOR SPEED (RPM) 287
FLIGHT CONDITION LEVEL
NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. DASE ON, ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION
STABILATOR POSITION (DEG) LED LEU
10
0
10
PITCH ATTITUDE (DEGREES) DN UP
10
0
10
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.06 INCHES
DIRECTIONAL CONTROL POSN (IN FROM LT) LT RT
5
4
3
2
TOTAL LATERAL CONTROL TRAVEL = 8.45 INCHES
LATERAL CONTROL POSITION (IN FROM LEFT) LT RT
6
5
4
3
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.5 INCHES
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD) FWD AFT
7
6
5
4
3
100
110
120
130
140
150
160
CALIBRATED AIRSPEED (KNOTS)

FIGURE 20

STATIC LONGITUDINAL STABILITY
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14780	208.5 (AFT)	7440	24.0	289	CLIMB

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. DASE ON, ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION

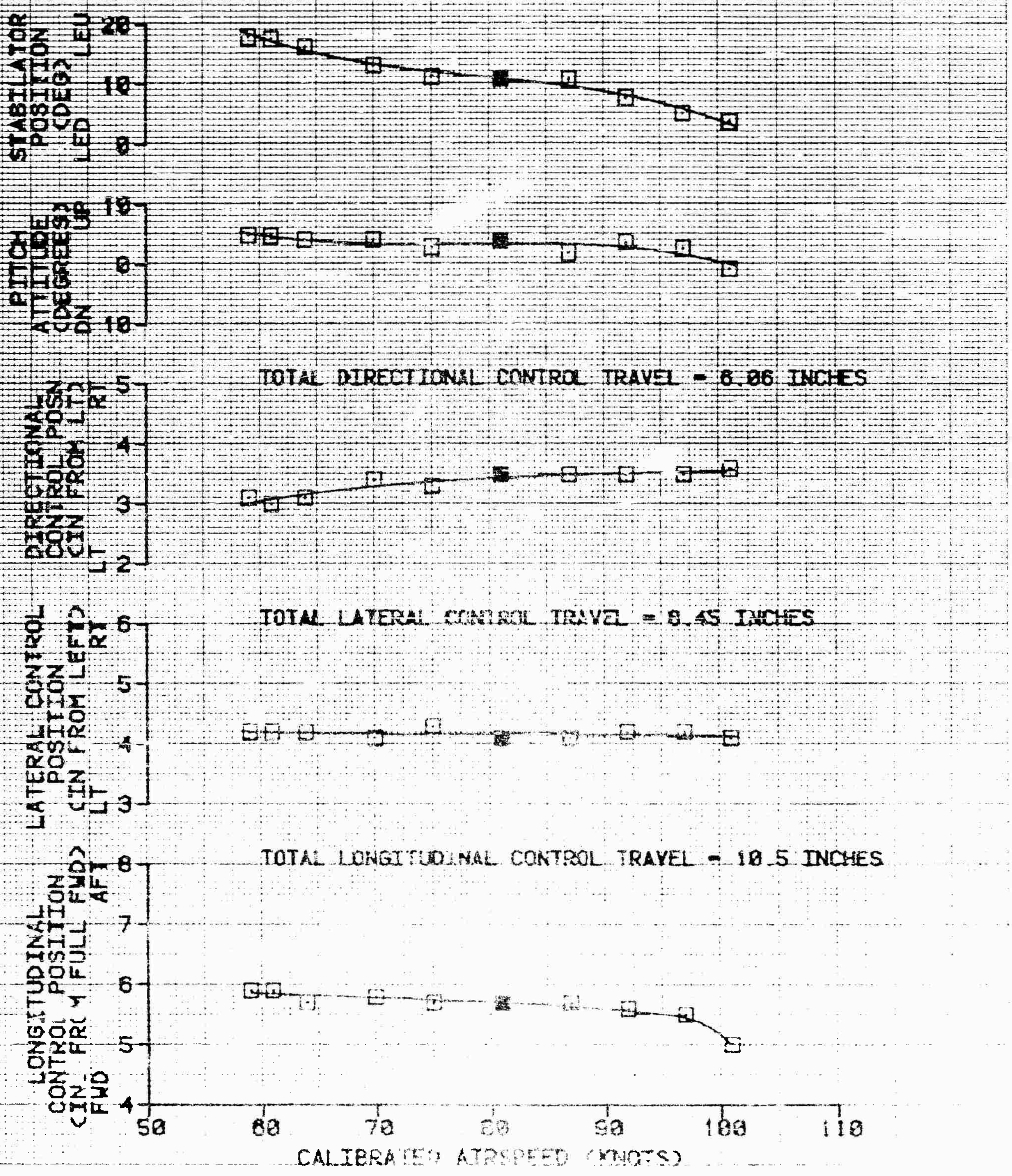


FIGURE 21
STATIC LONGITUDINAL STABILITY
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14780	200.5 (AFT)	7340	23.0	294	DESCENT

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. SASE ON, ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION

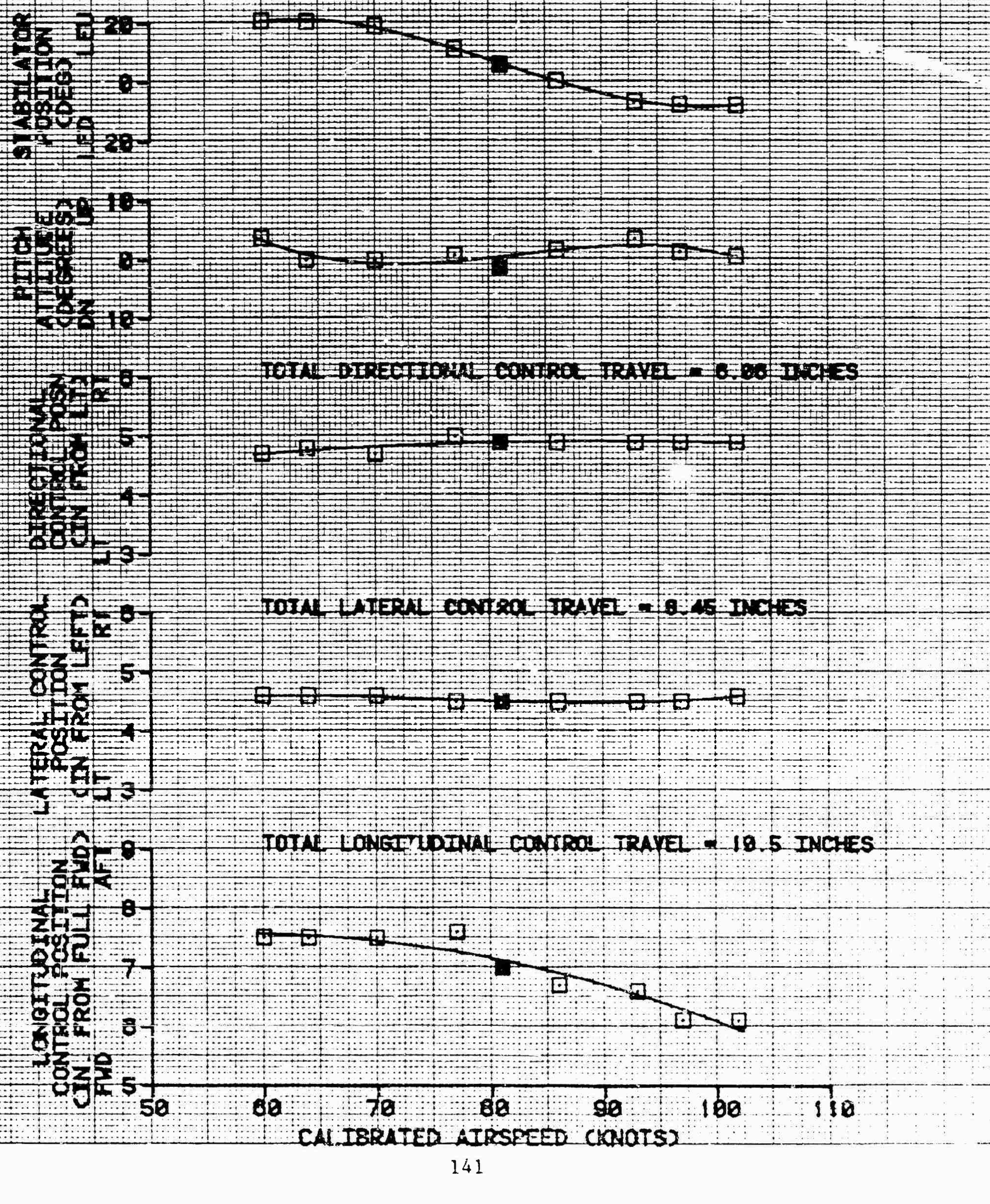


FIGURE 22

STATIC LATERAL-DIRECTIONAL STABILITY

AH-64A USA S/N 82-23355

AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
200.5 (AFT)	7250	24.0	290	70	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. DASE ON, ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION

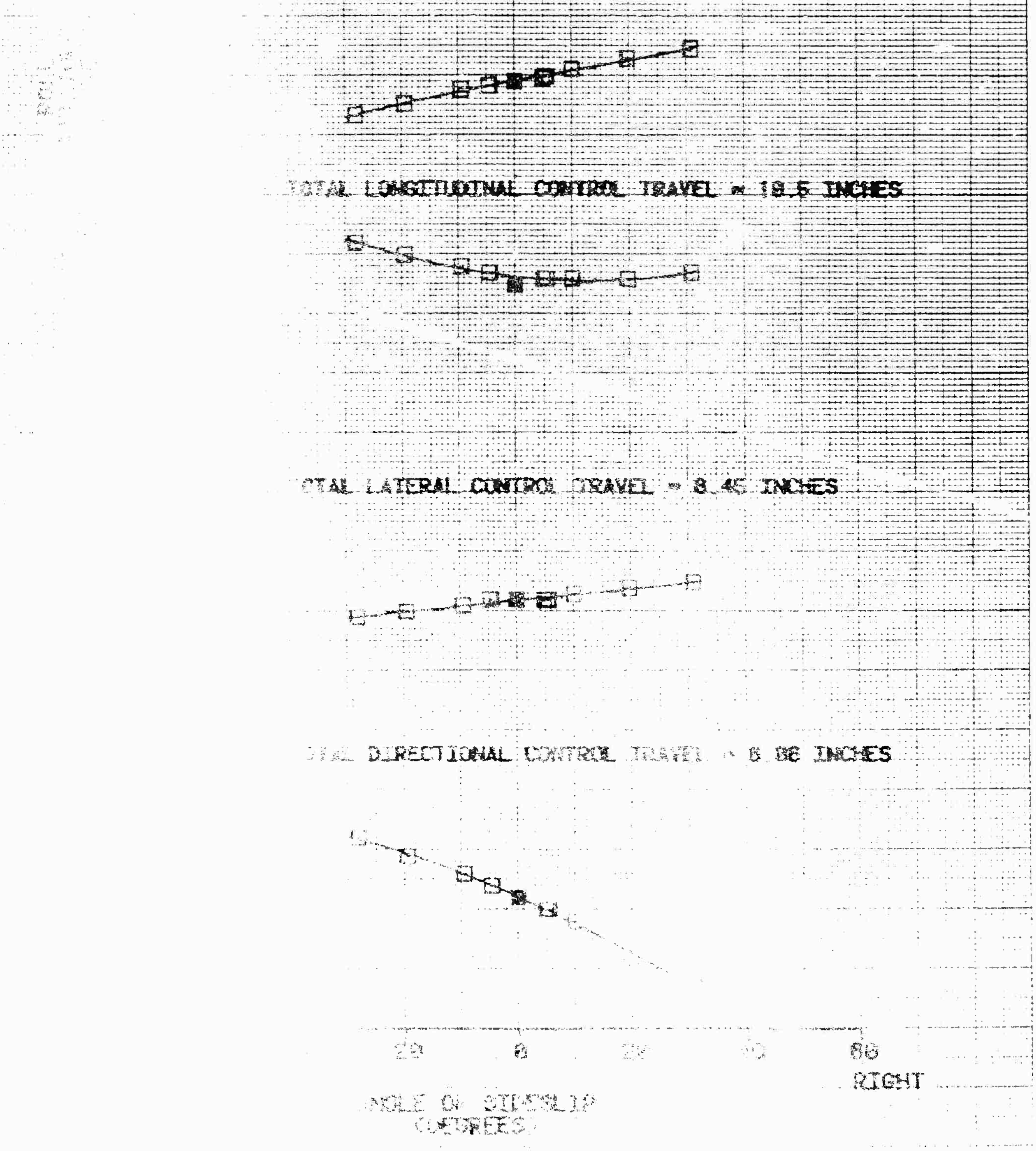


FIGURE 23

STATIC LATERAL-DIRECTIONAL STABILITY

AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
14570	200.7 (AFT)	7050	20.0	287	130	LEVEL

NOTES: 1. SHADED SYMBOLS DENOTE TRIM

2. DASE ON, ATTITUDE HOLD OFF

3. 8-HELLFIRE CONFIGURATION

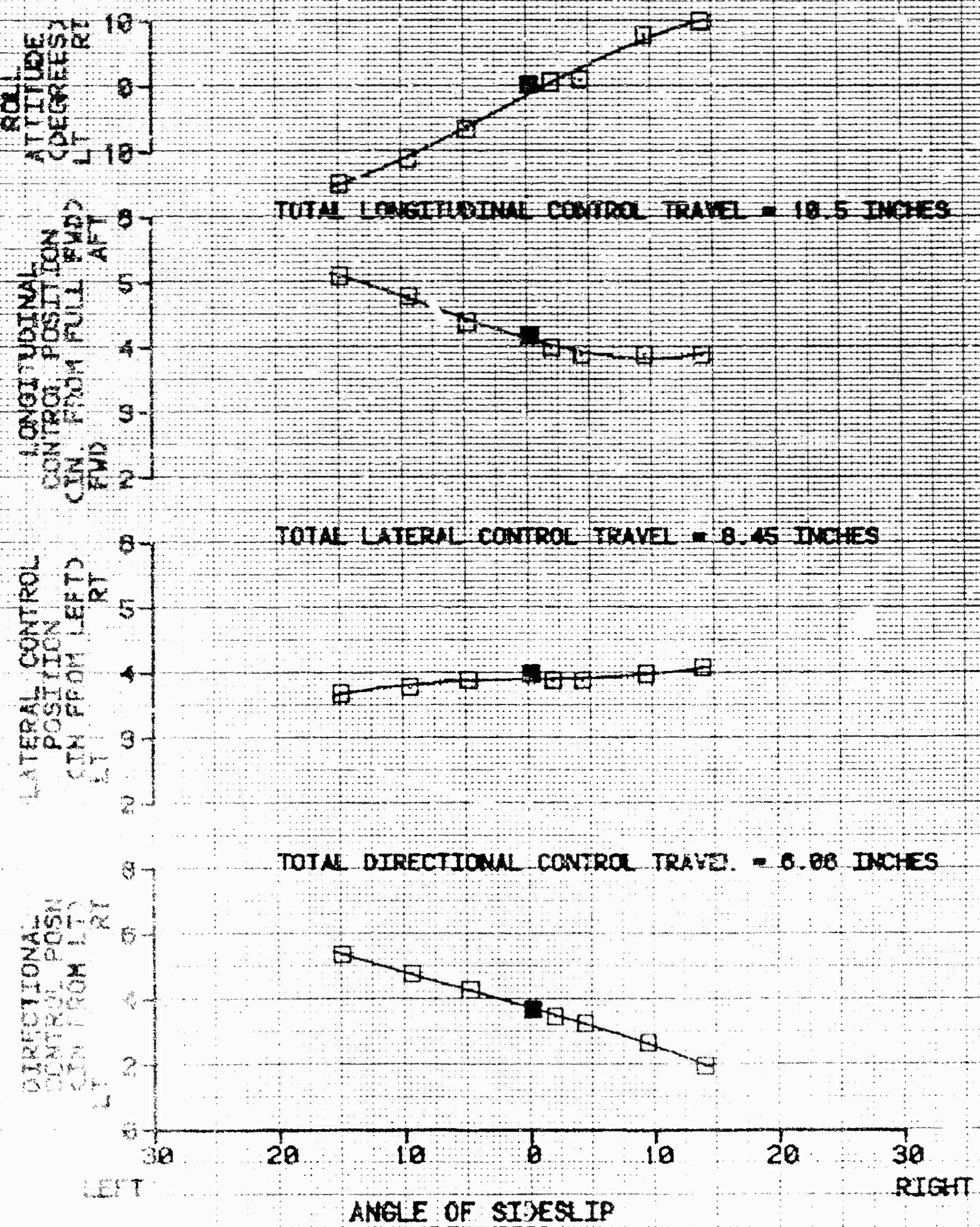


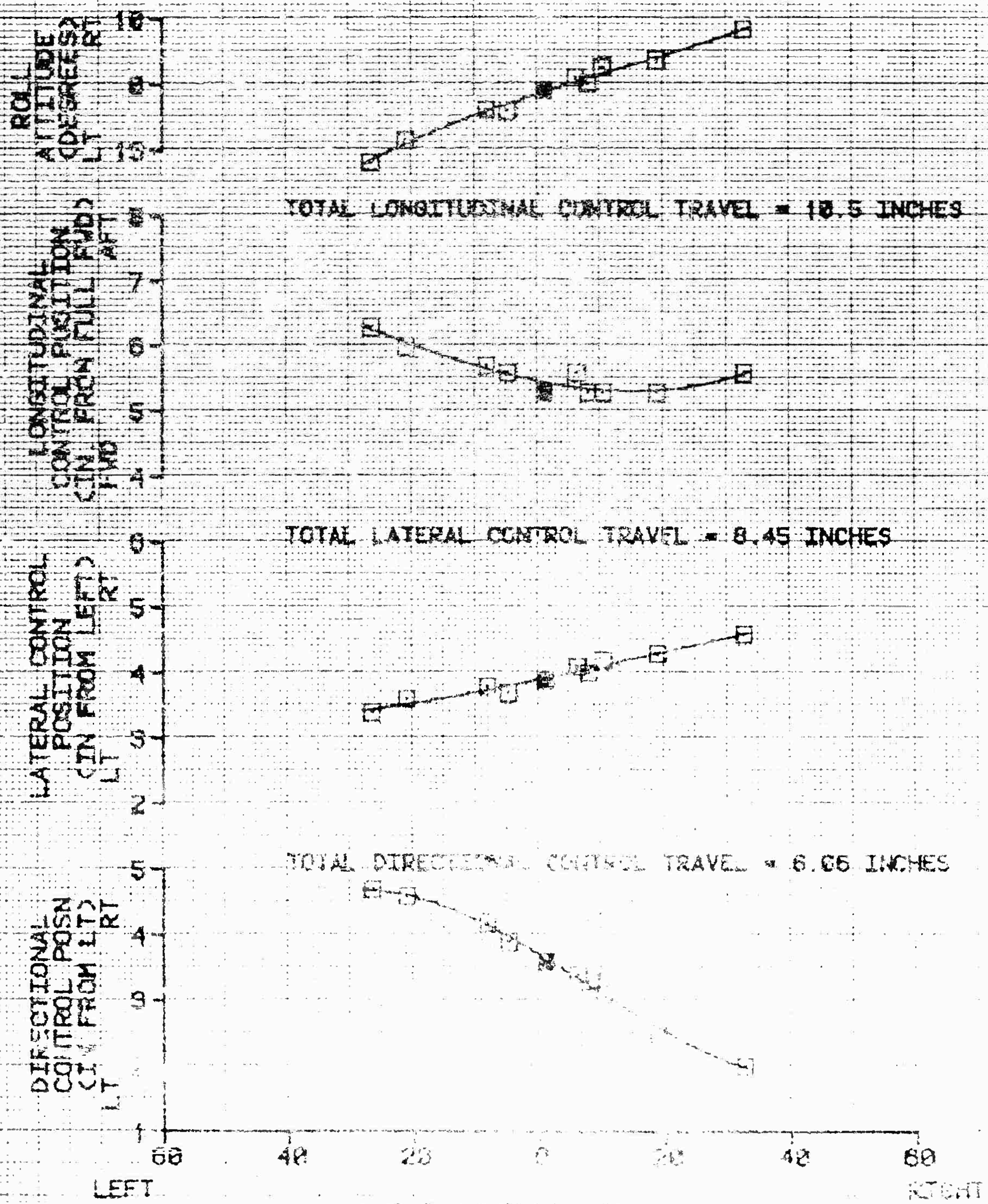
FIGURE 24

STATIC LATERAL-DIRECTIONAL STABILITY

AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
14670	200.9 (AFT)	7250	21.5	289	81	CLIMB

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. BASE SCL ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION



ANGLE OF SIDESLIP (DEGREES)

FIGURE 25
STATIC LATERAL-DIRECTIONAL STABILITY
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
14670	200.3 (AFT)	6030	21.5	289	80	DESCENT

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. DASE ON, ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION

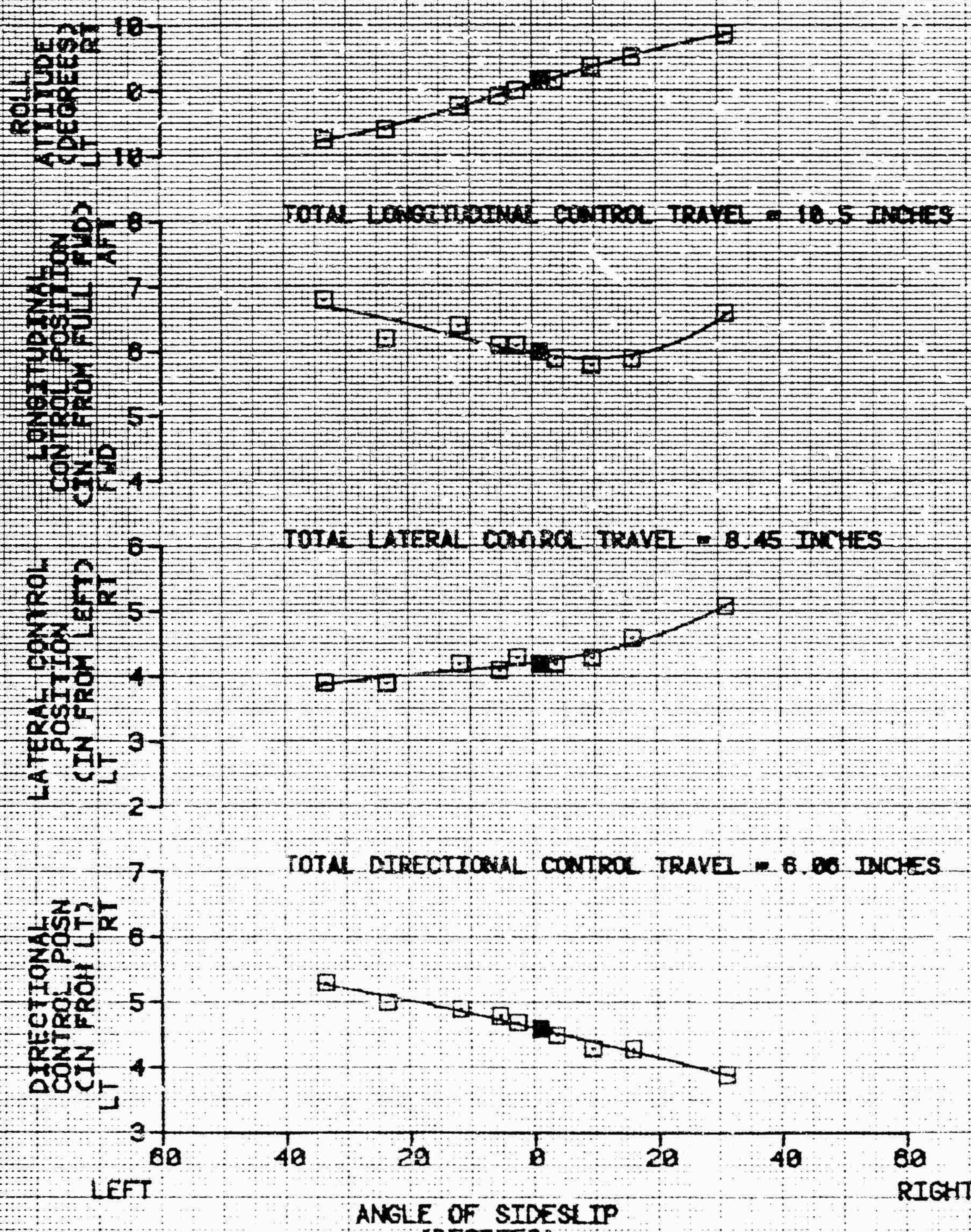


FIGURE 26
MANEUVERING STABILITY
AH-64A USA S/N 82-23355

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIB. AIRSPEED (KNOTS)
14800	206.1 (AFT)	7170	23.0	288	135

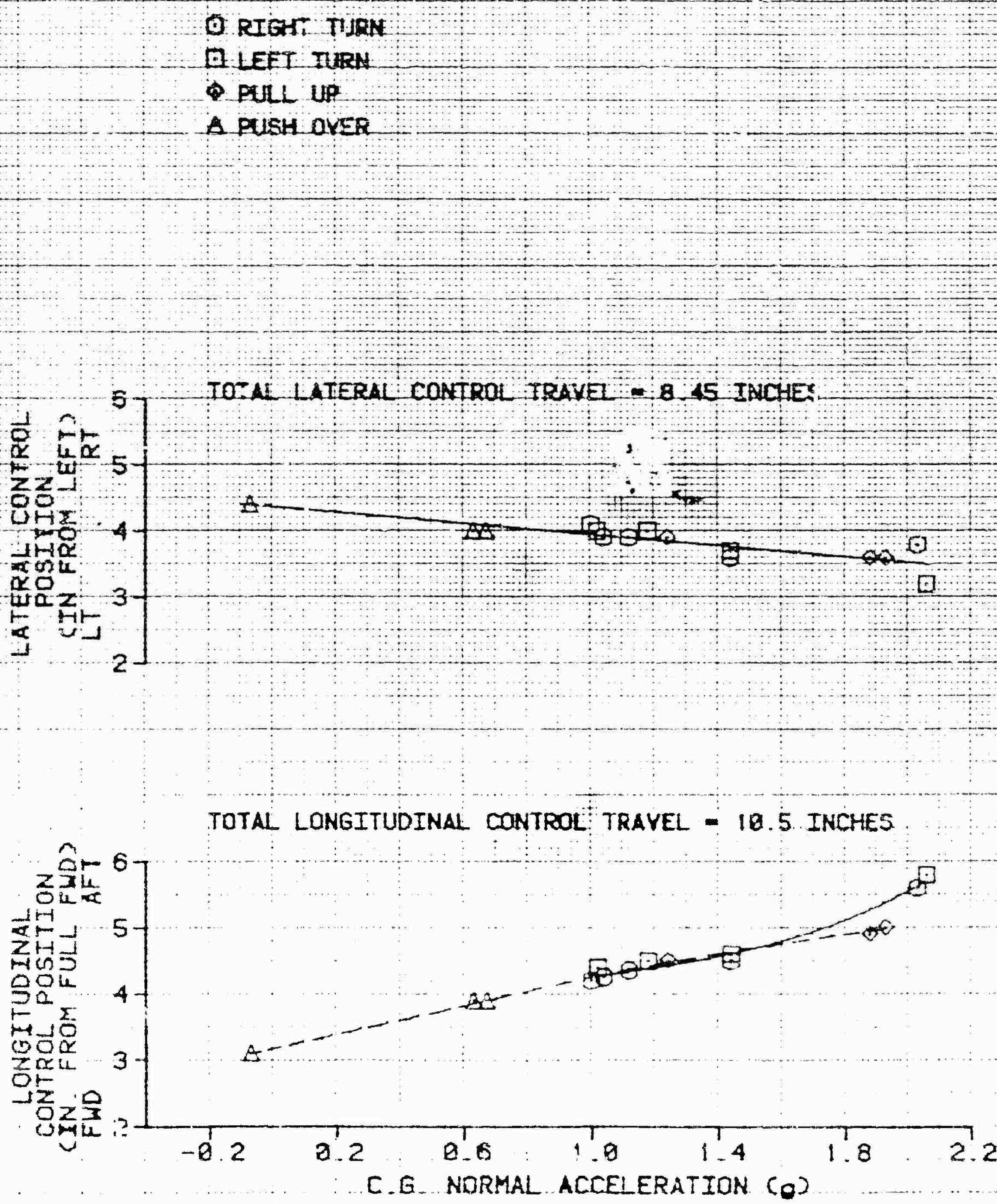


FIGURE 27
LONG TERM DYNAMIC STABILITY
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14690	206.4 (AFT)	7370	29.0	290	LEVEL

NOTE: DASE ON ATTITUDE HOLD ON

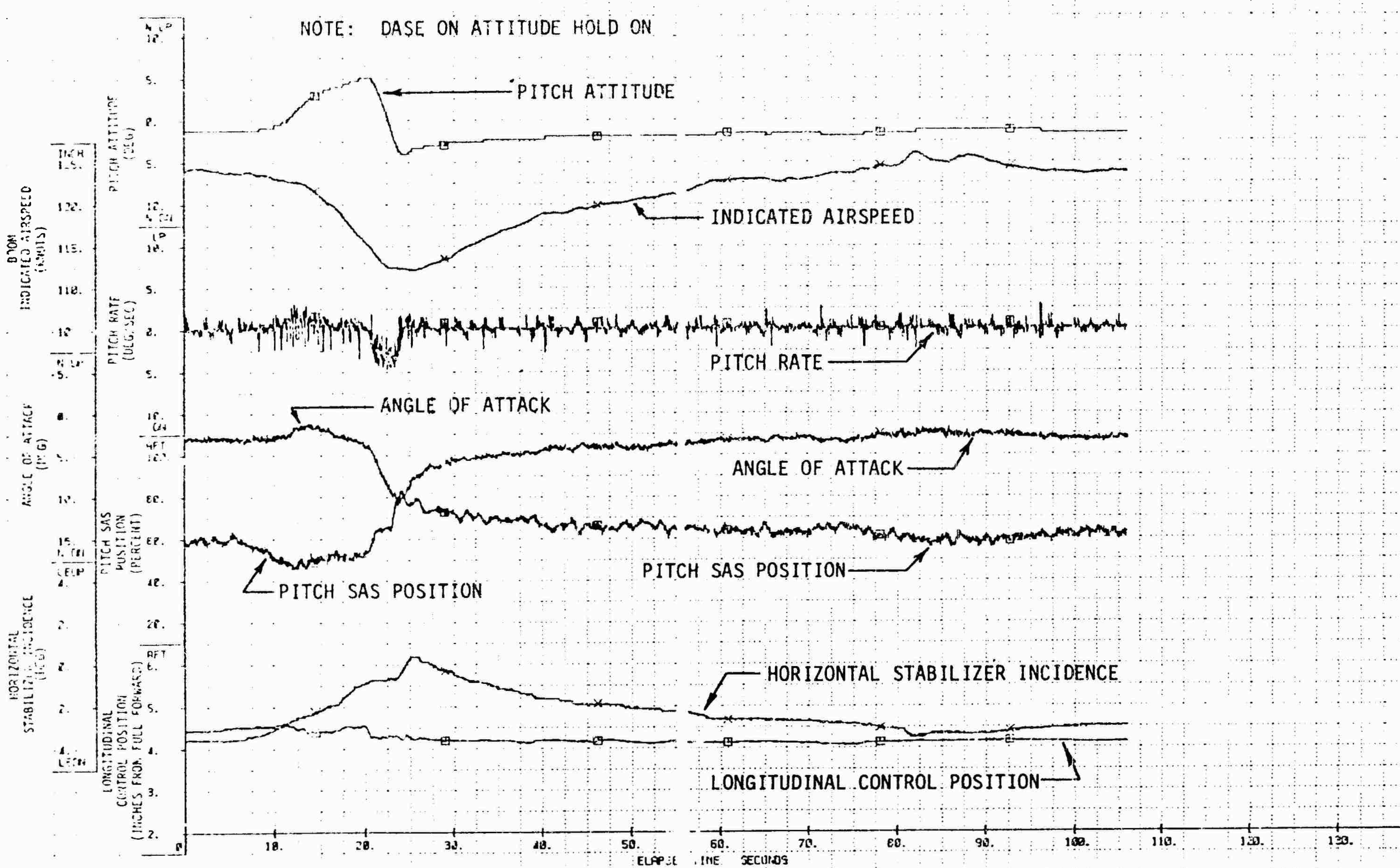


FIGURE 28
LONG TERM DYNAMIC STABILITY
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14690	206.4 (AFT)	7030	29.0	290	LEVEL

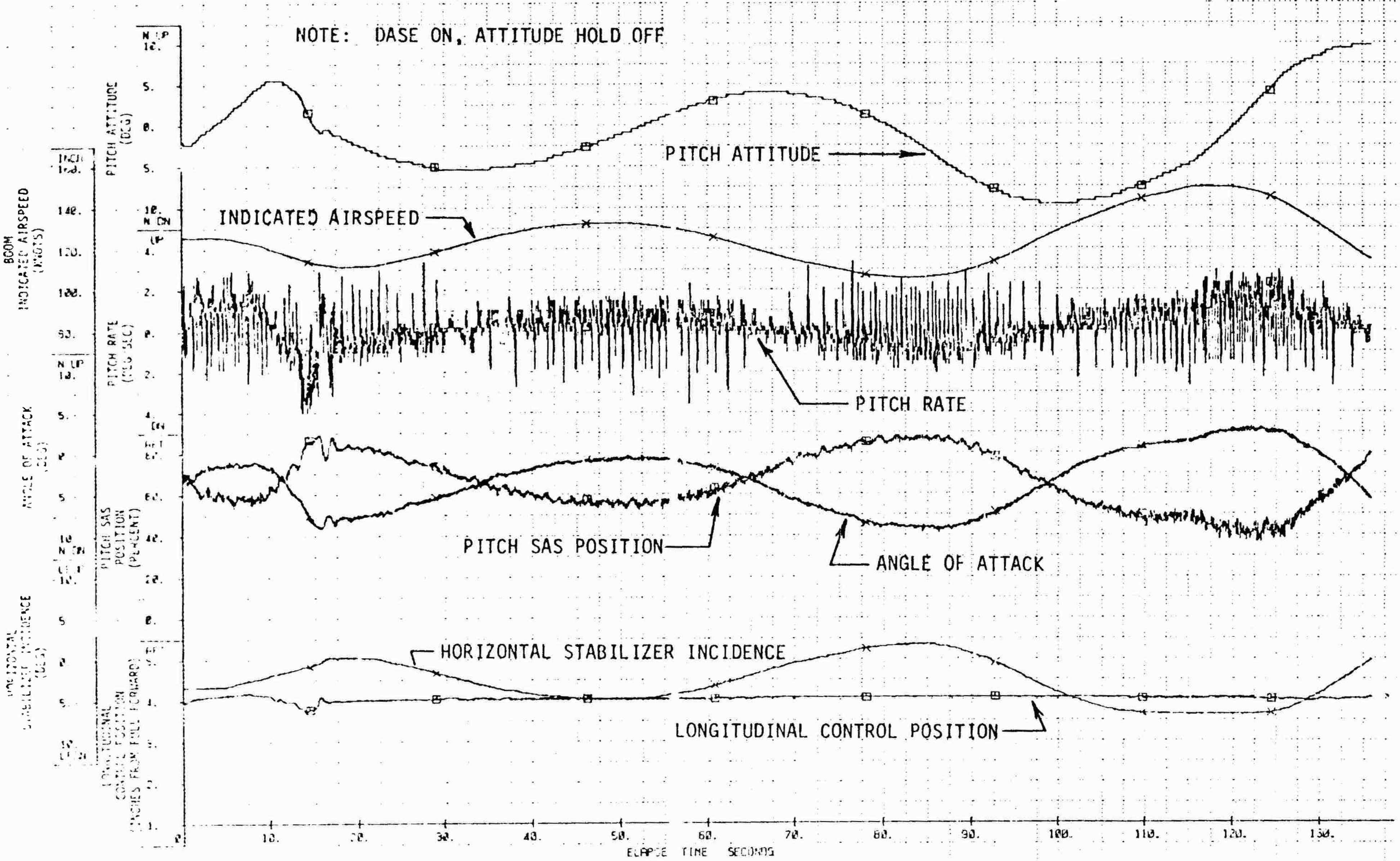


FIGURE 29
DYNAMIC STABILITY
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14690	206.4 (AFT)	6820	29.5	290	LEVEL

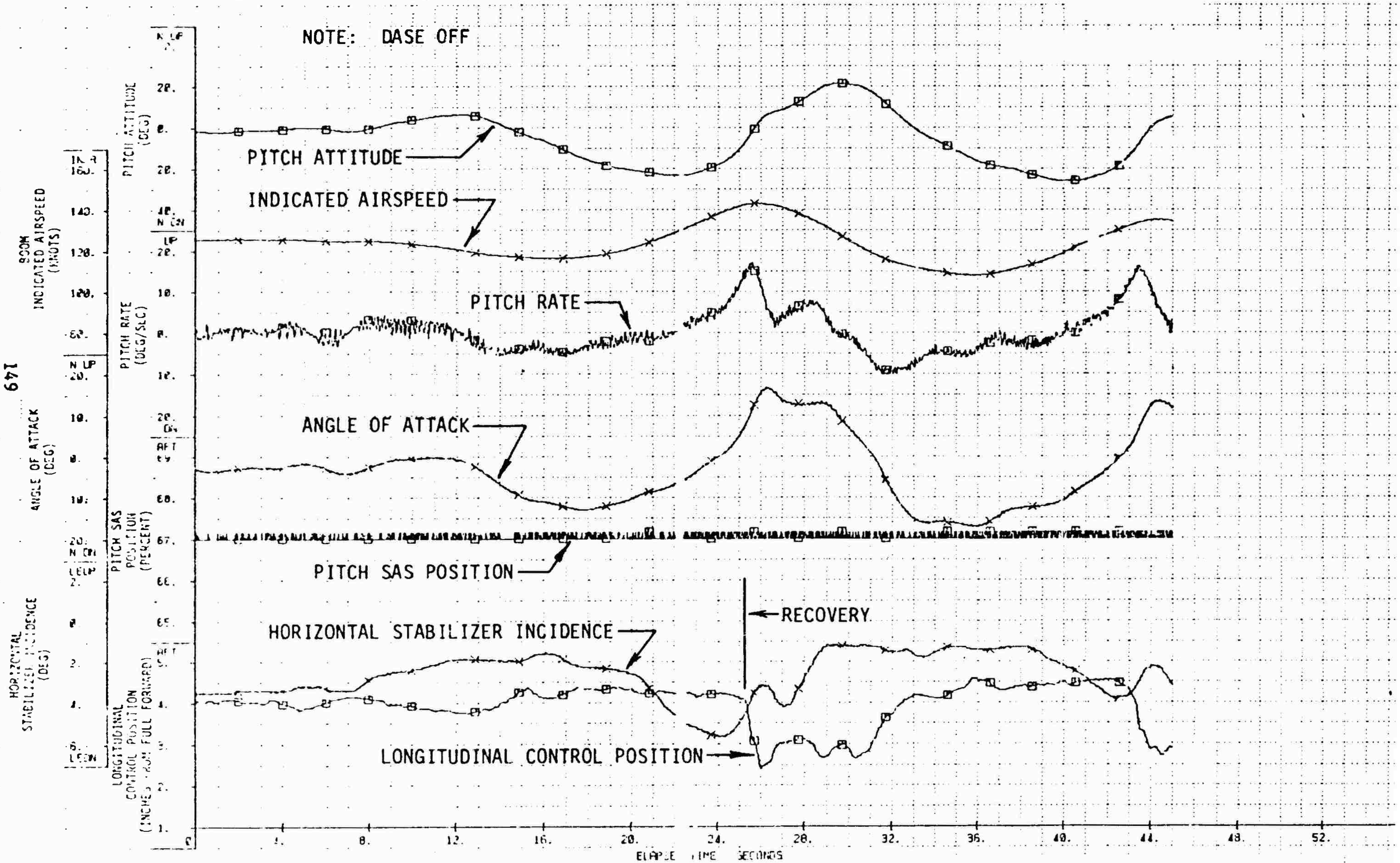


FIGURE 30
DYNAMIC STABILITY
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14690	206.4 (AFT)	6990	28.0	290	LEVEL

NOTE: DASE ON, ATTITUDE HOLD ON

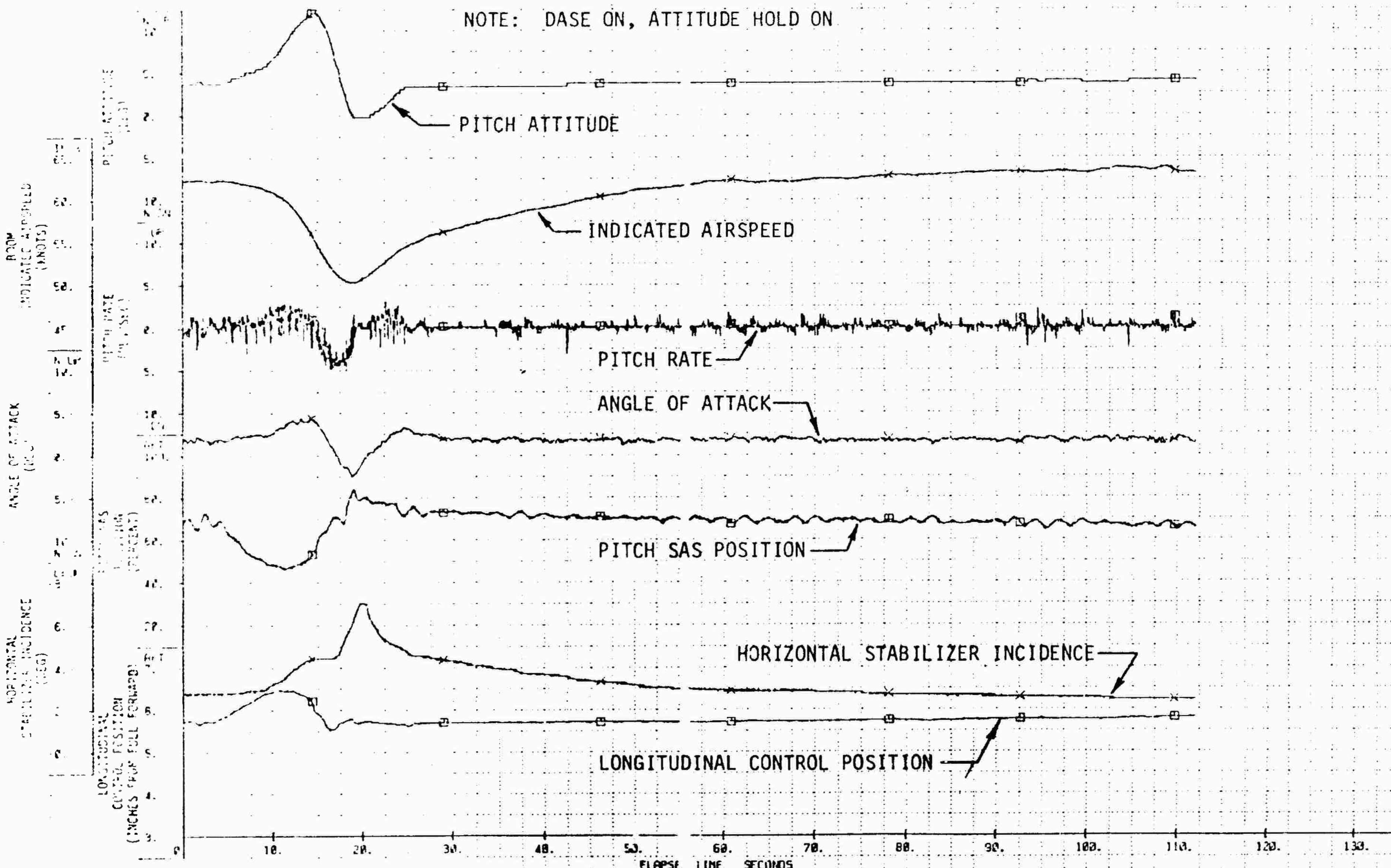


FIGURE 31
DYNAMIC STABILITY
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14690	206.4 (AFT)	6820	29.0	290	LEVEL

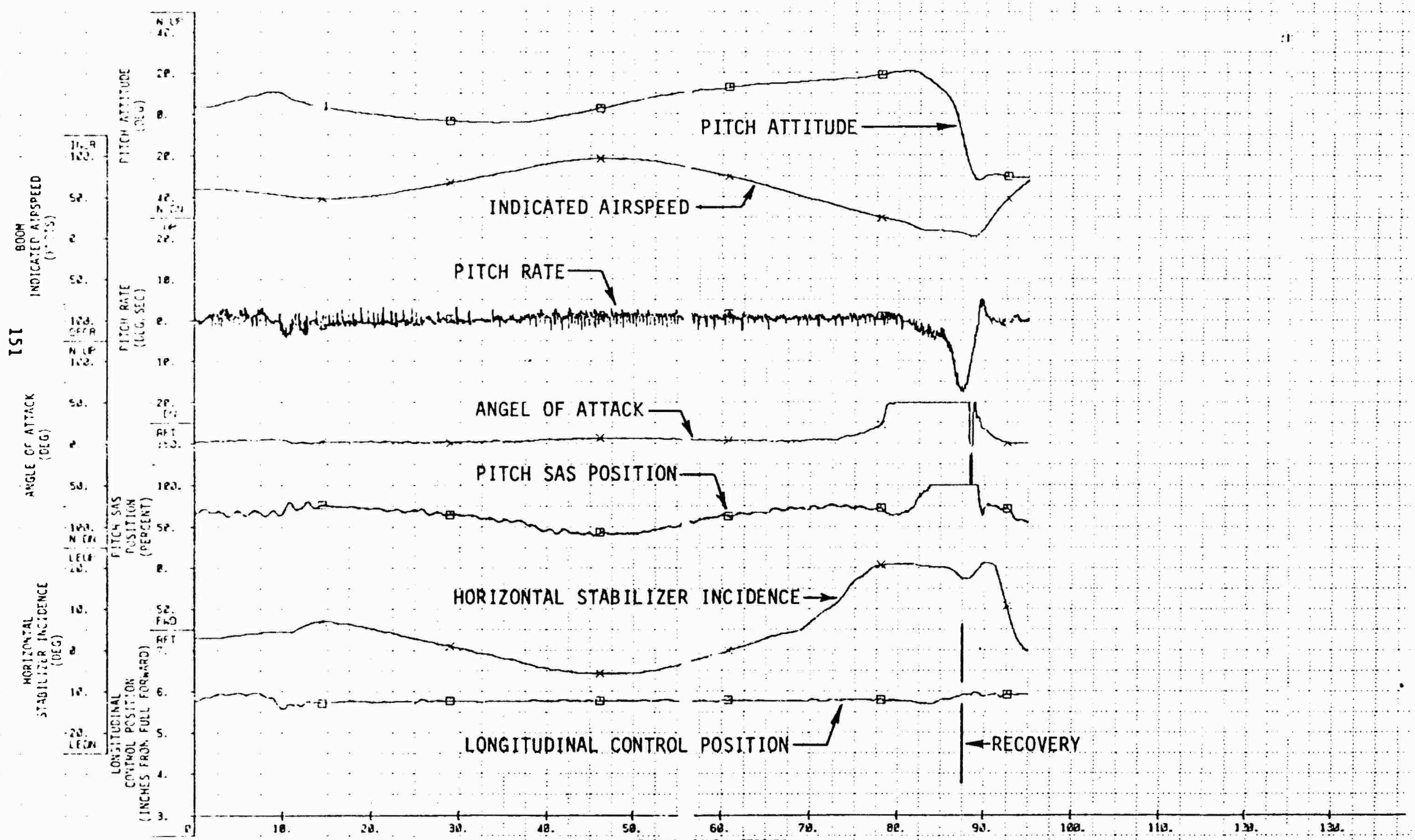


FIGURE 32
CONTROL SYSTEM IMBALANCE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14770	206.4 (AFT)	7620	24.0	290	LEVEL

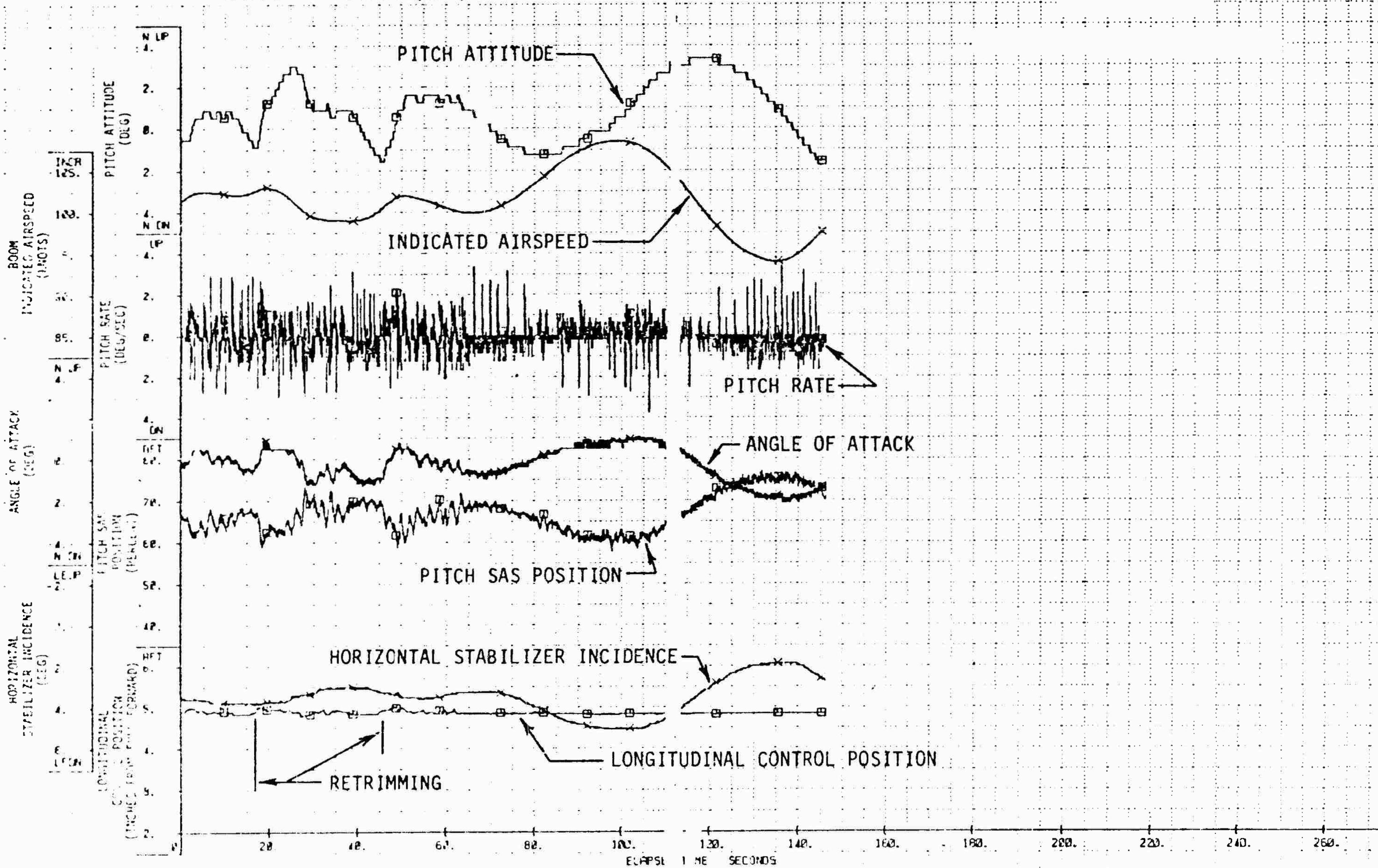


FIGURE 33
AFT LONGITUDINAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.7 (AFT)	6960	27.0	290	LEVEL

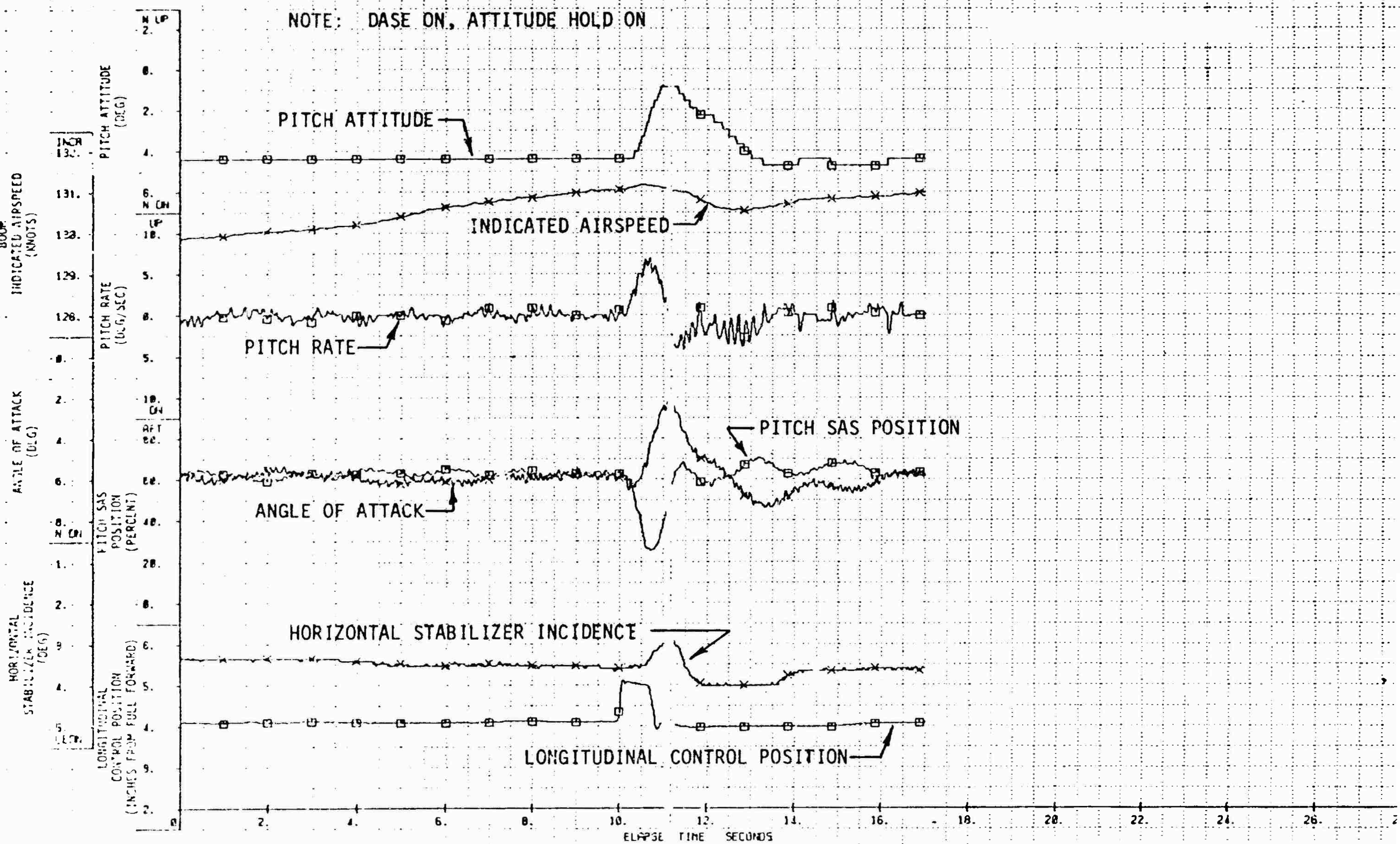


FIGURE 34
AFT LONGITUDINAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.7 (AFT)	7210	26.1	290	LEVEL

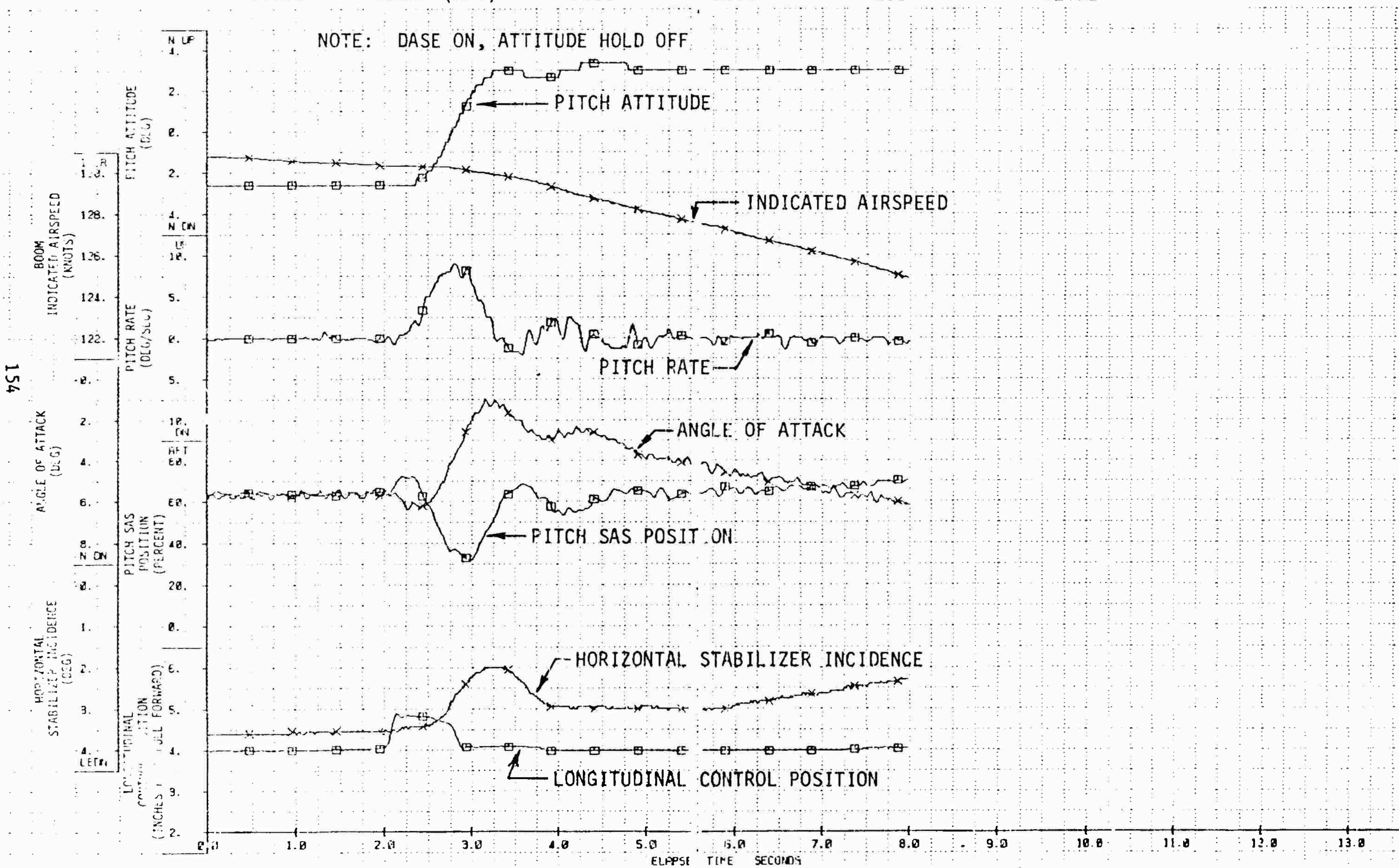


FIGURE 35
FORWARD LONGITUDINAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.4 (AFT)	7110	21.5	290	LEVEL

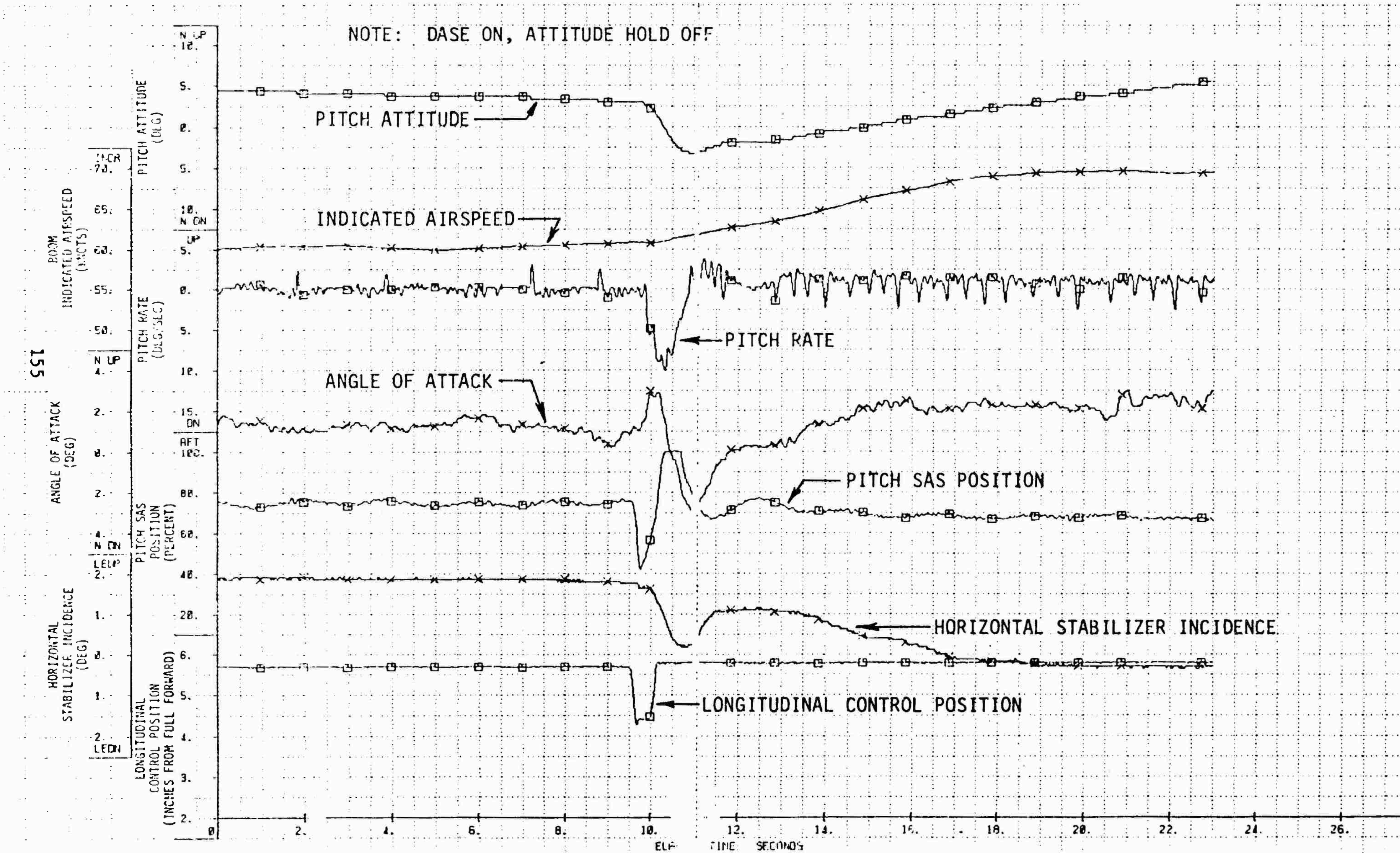


FIGURE 36A
FORWARD LONGITUDINAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.6 (AFT)	7230	20.0	290	LEVEL

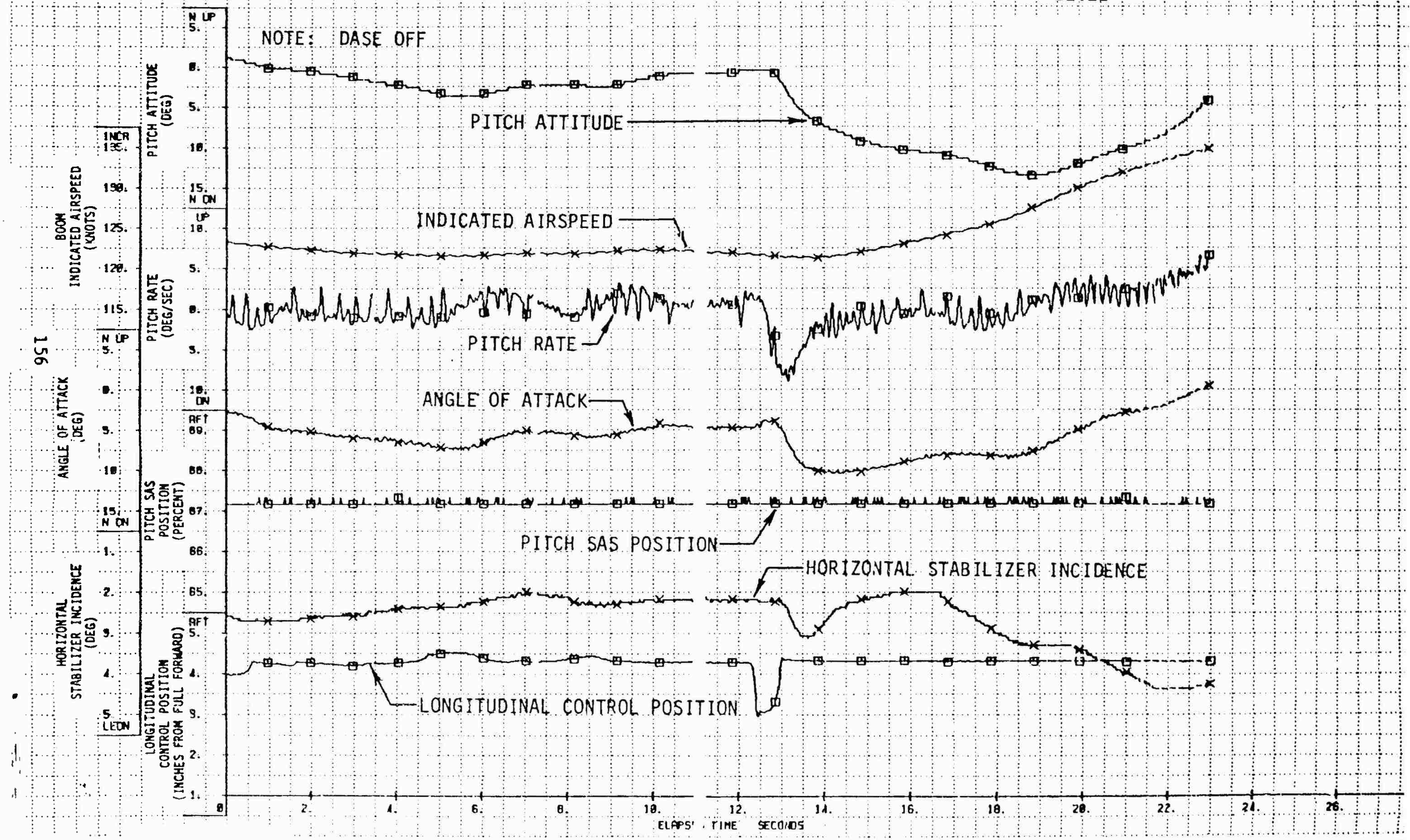


FIGURE 36B
FORWARD LONGITUDINAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.6 (AFT)	7230	20.0	290	LEVEL

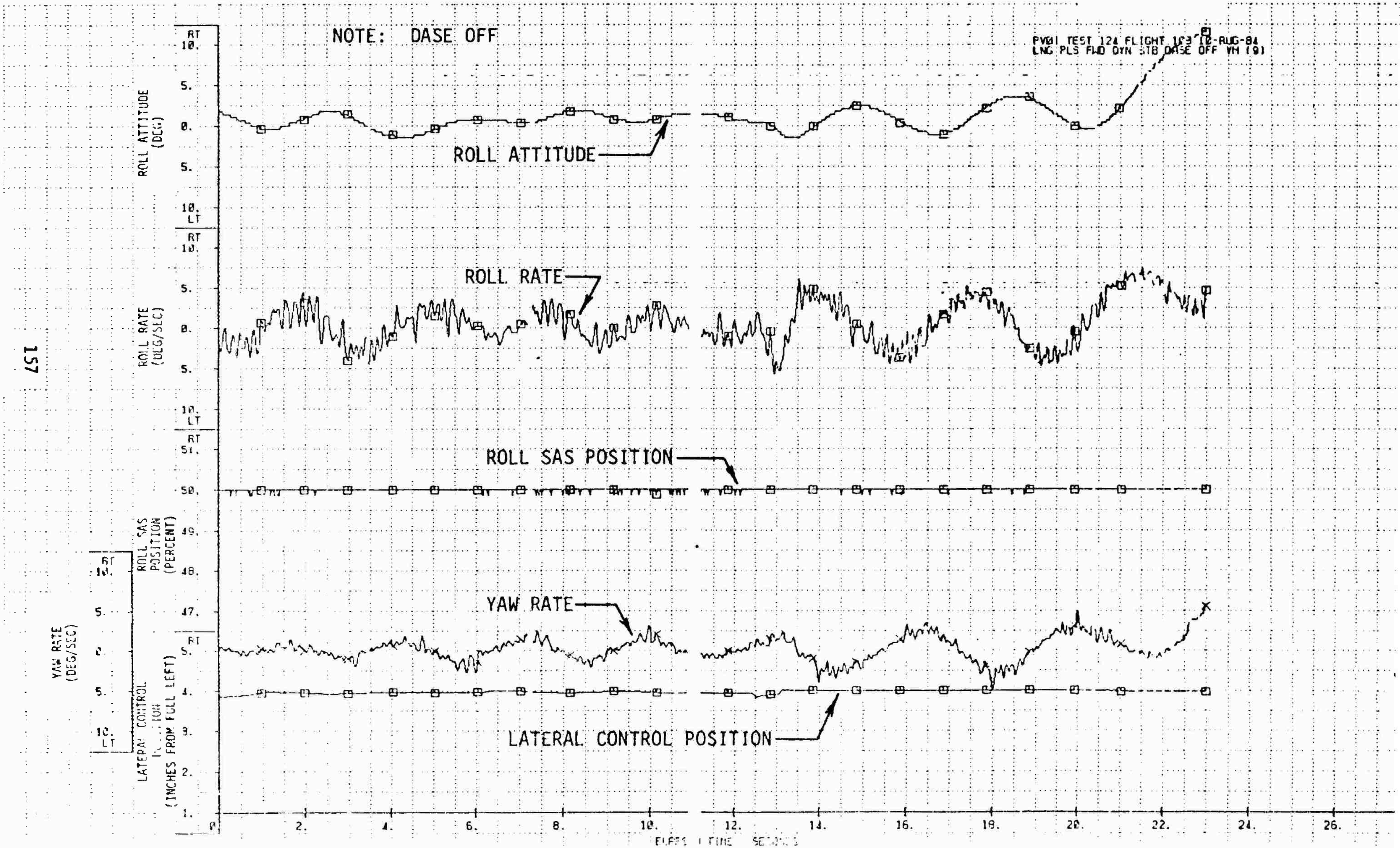


FIGURE 37A
RIGHT LATERAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.7 (AFT)	7010	26.5	290	LEVEL

NOTE: DASE ON, ATTITUDE HOLD OFF

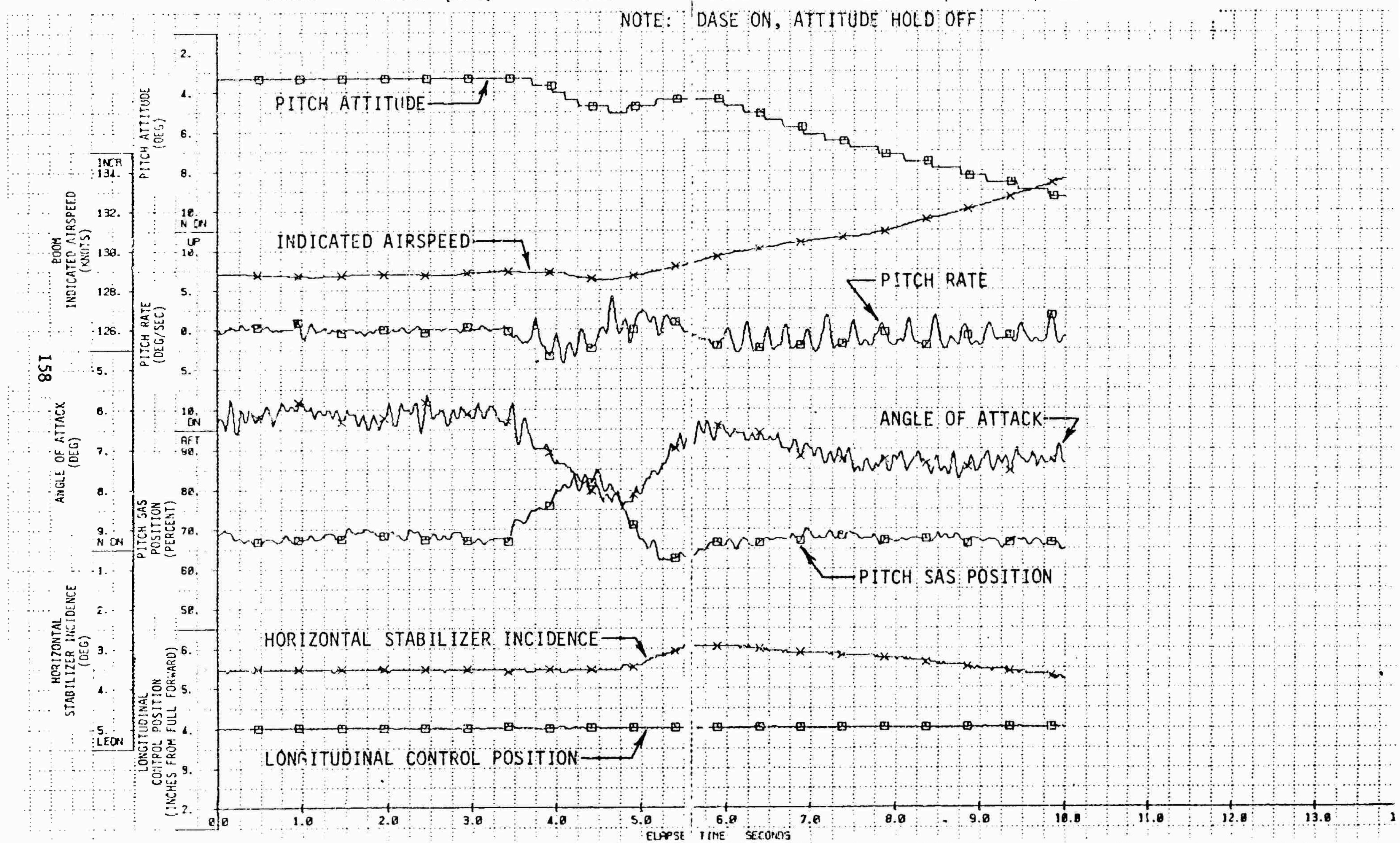


FIGURE 37B
RIGHT LATERAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.7 (AFT)	7010	26.5	290	LEVEL

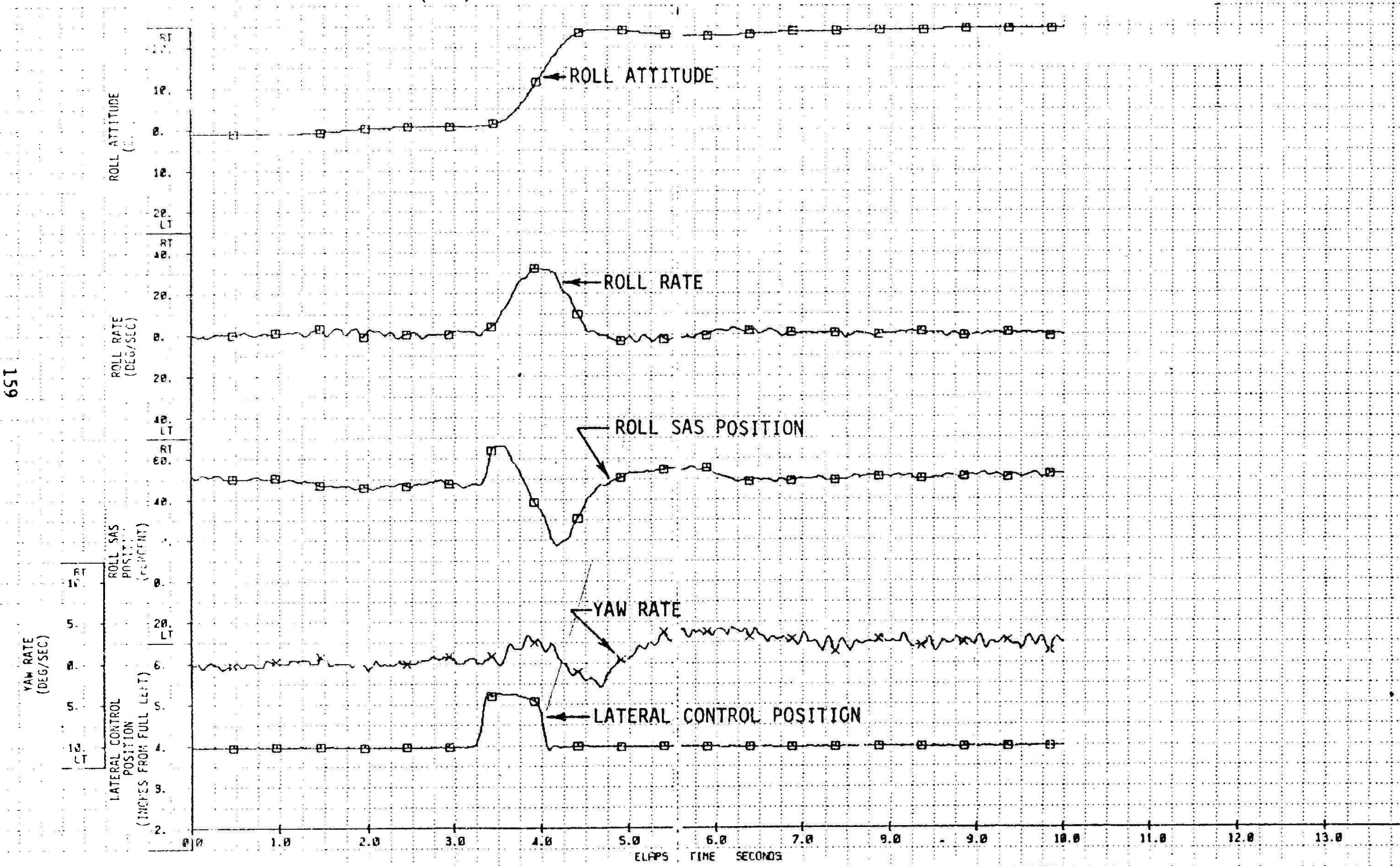


FIGURE 38A
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.6 (AFT)	7390	20.5	290	LEVEL

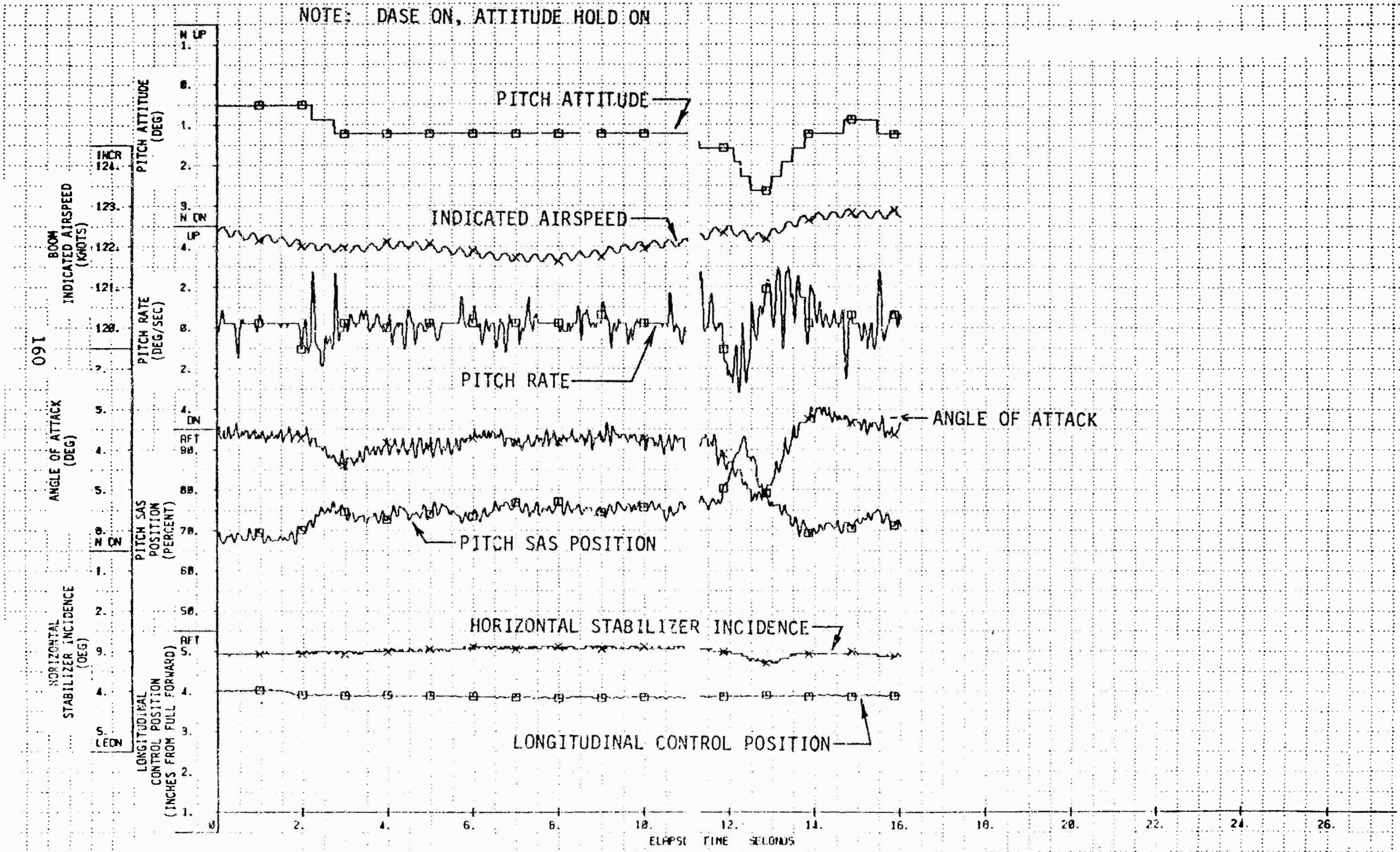


FIGURE 38B
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.6 (AFT)	7390	20.5	290	LEVEL

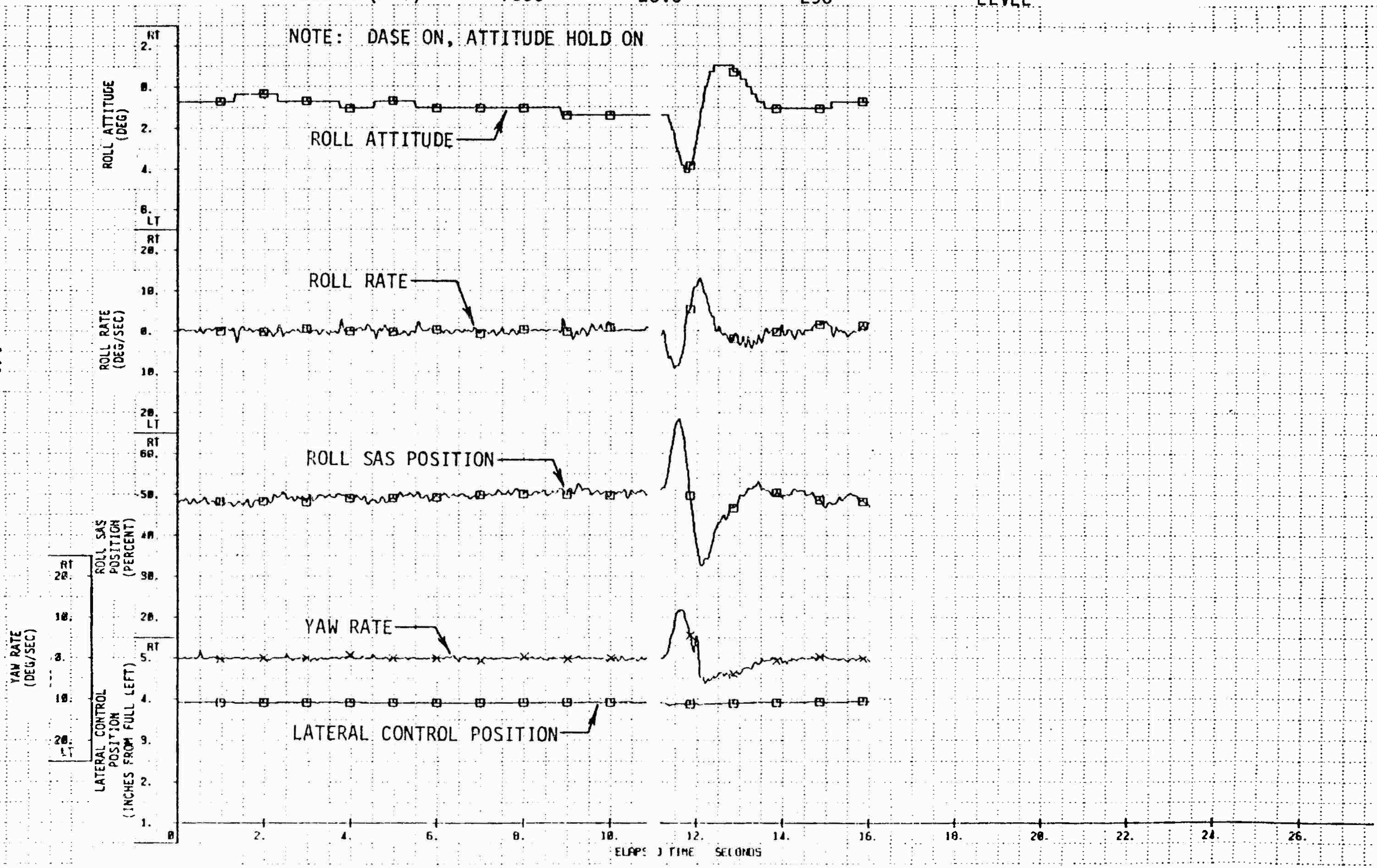


FIGURE 38C
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.6 (AFT)	7390	20.5	290	LEVEL

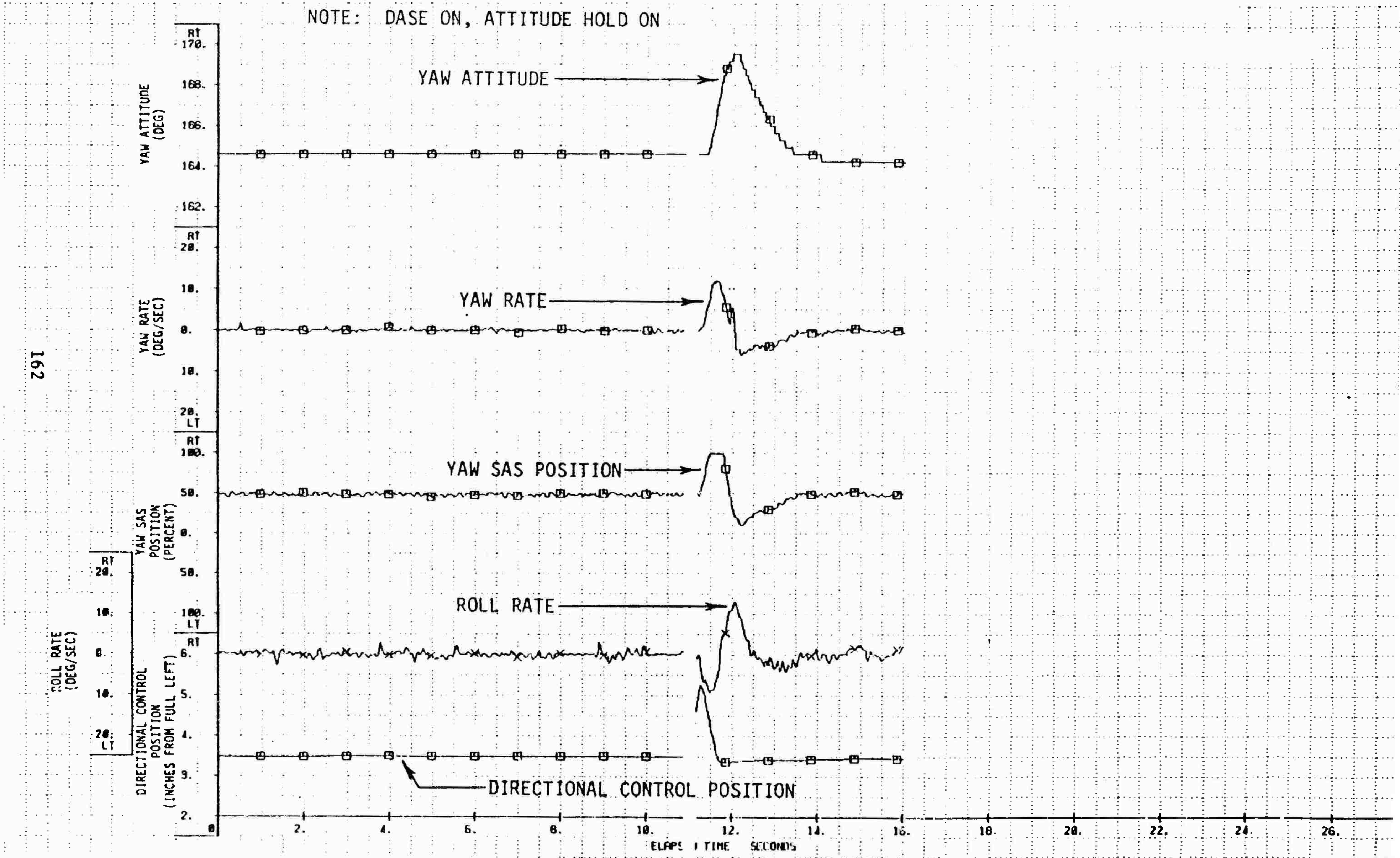


FIGURE 39A
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.4 (AFT)	6950	22.0	290	LEVEL

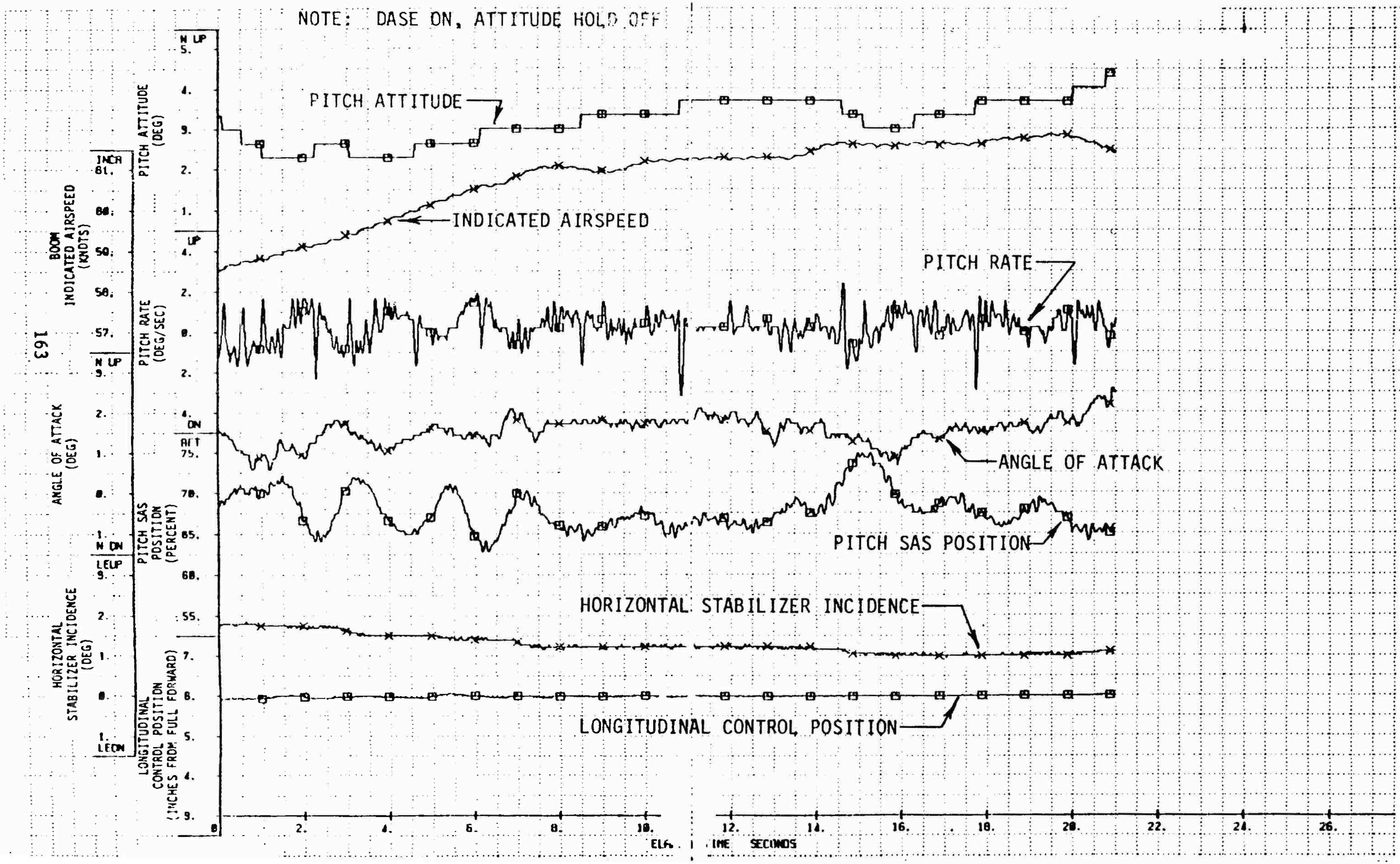


FIGURE 39B
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.4 (AFT)	6950	22.0	290	LEVEL

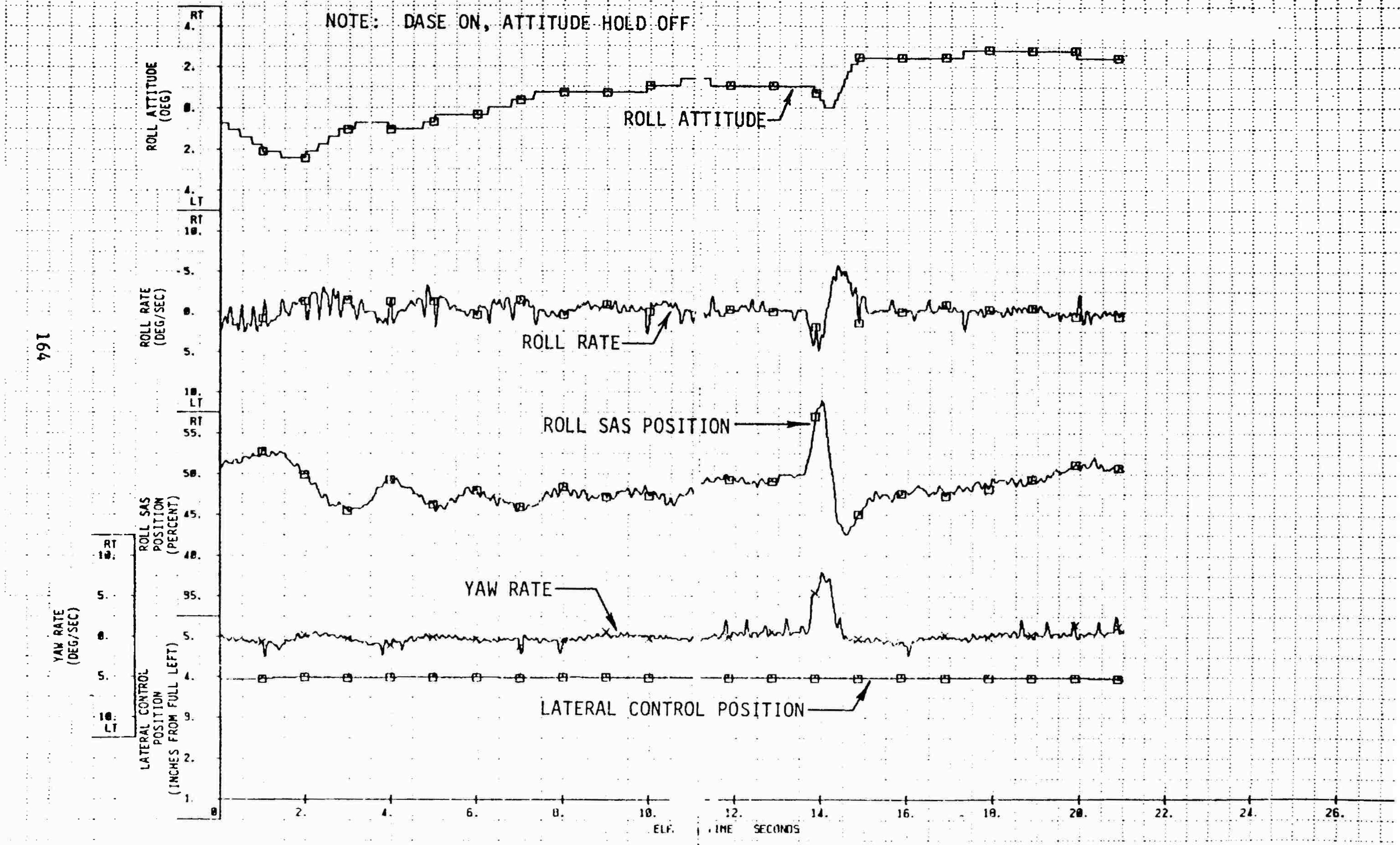


FIGURE 39C
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14650	206.4 (AFT)	6950	22.0	290	LEVEL

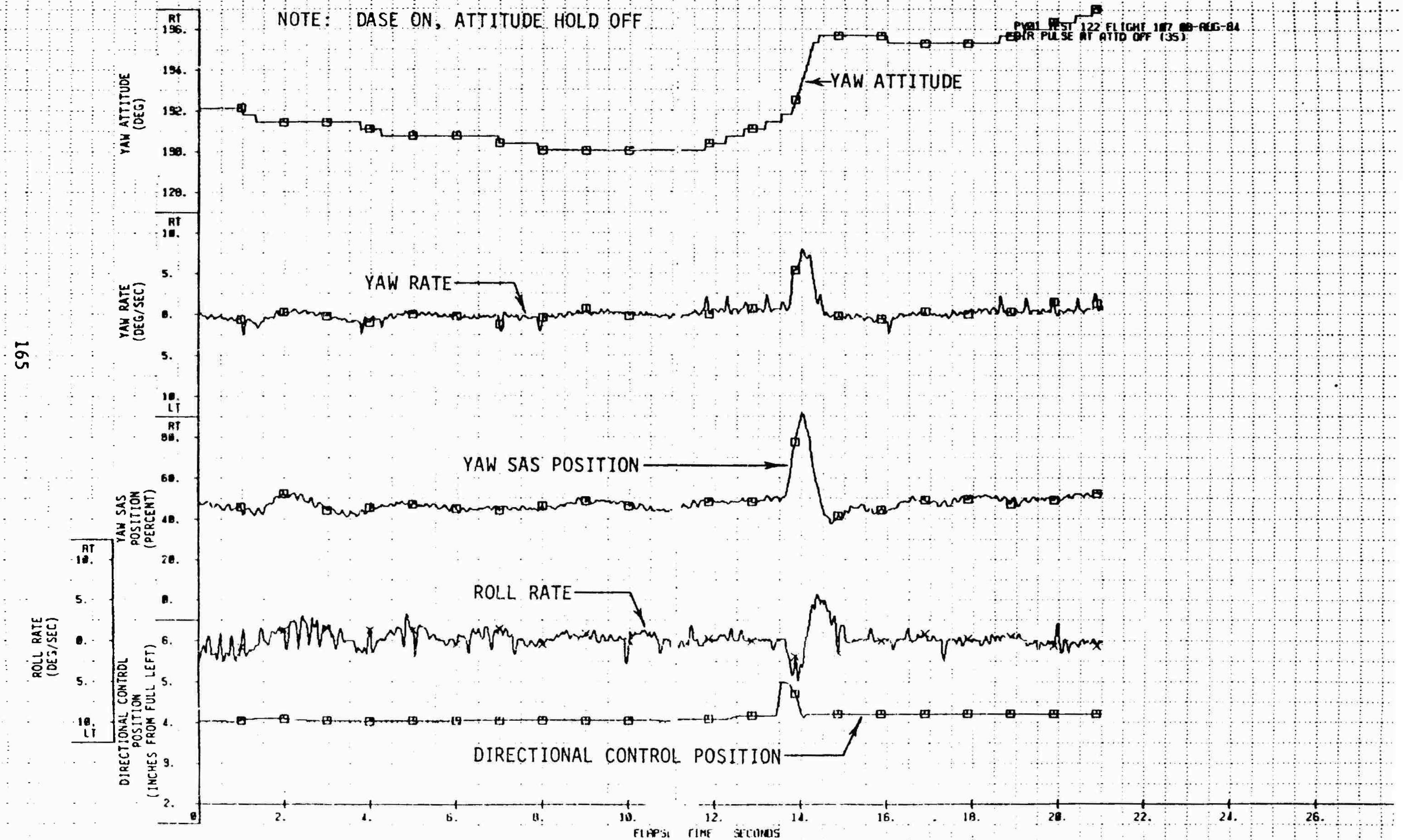


FIGURE 40A
LEFT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.6 (AFT)	7270	21.0	290	LEVEL

NOTE: DASE OFF

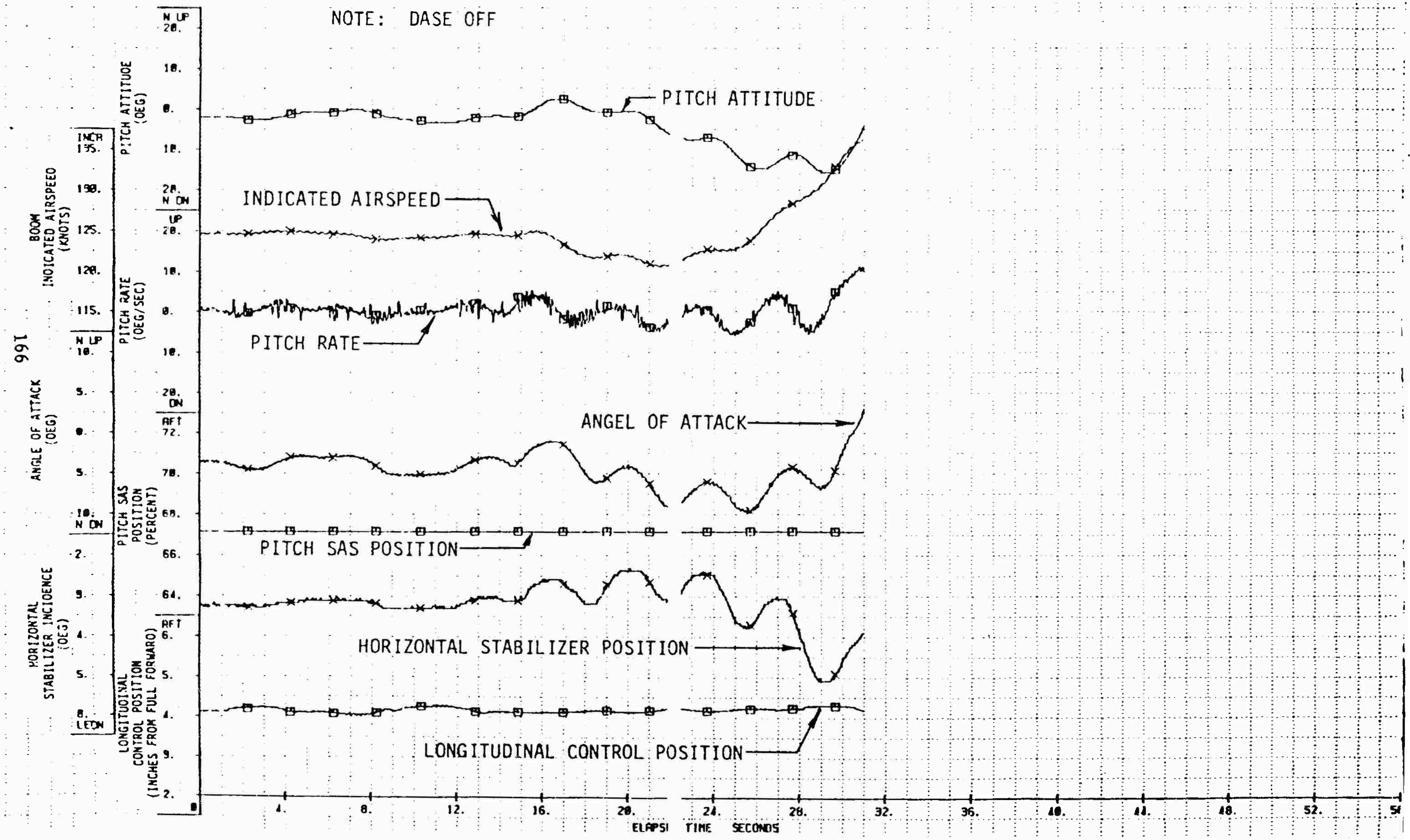


FIGURE 40B
LEFT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.6 (AFT)	7270	21.0	290	LEVEL

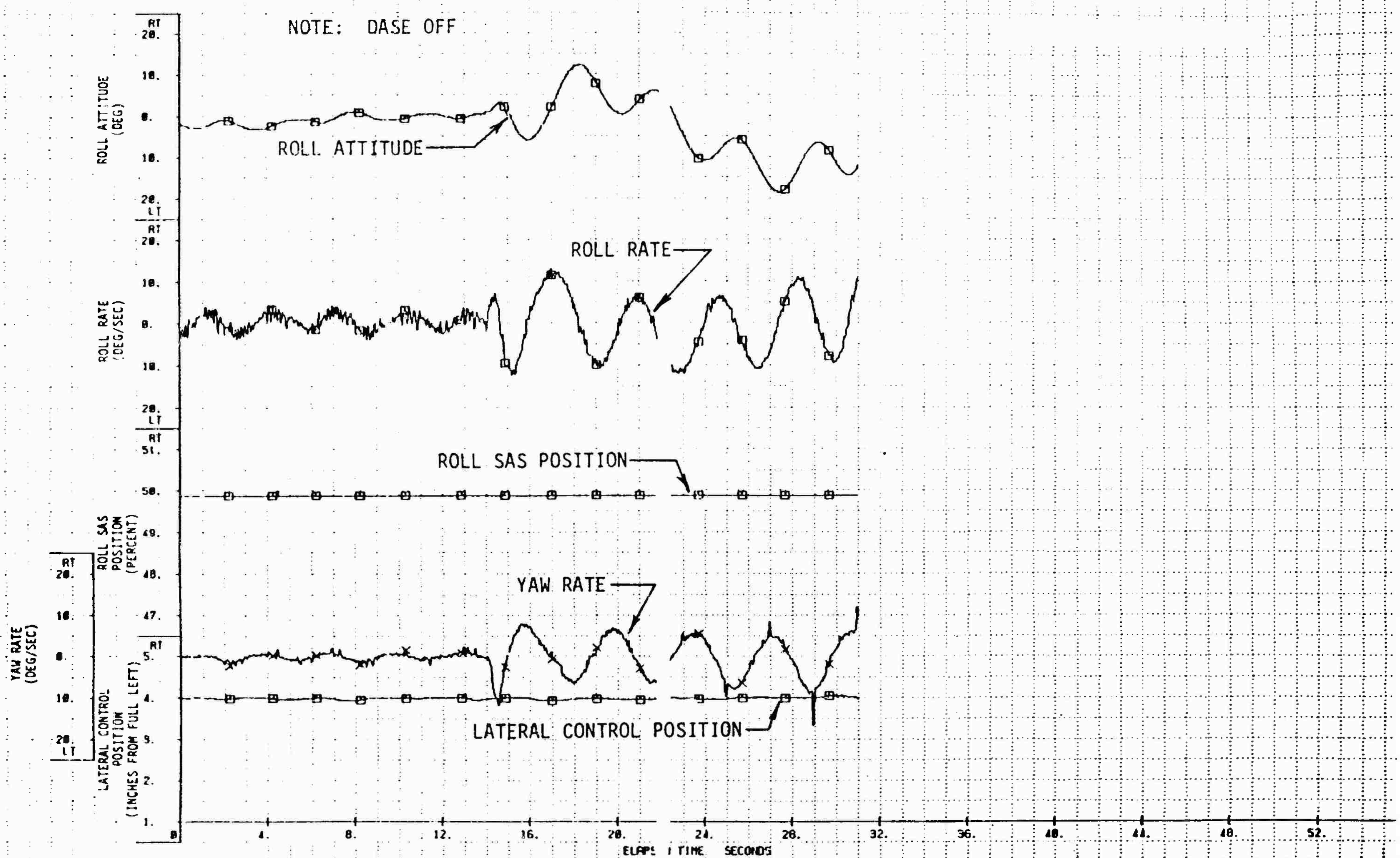


FIGURE 40C
LEFT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
14760	206.5 (AFT)	7270	21.0	290	LEVEL

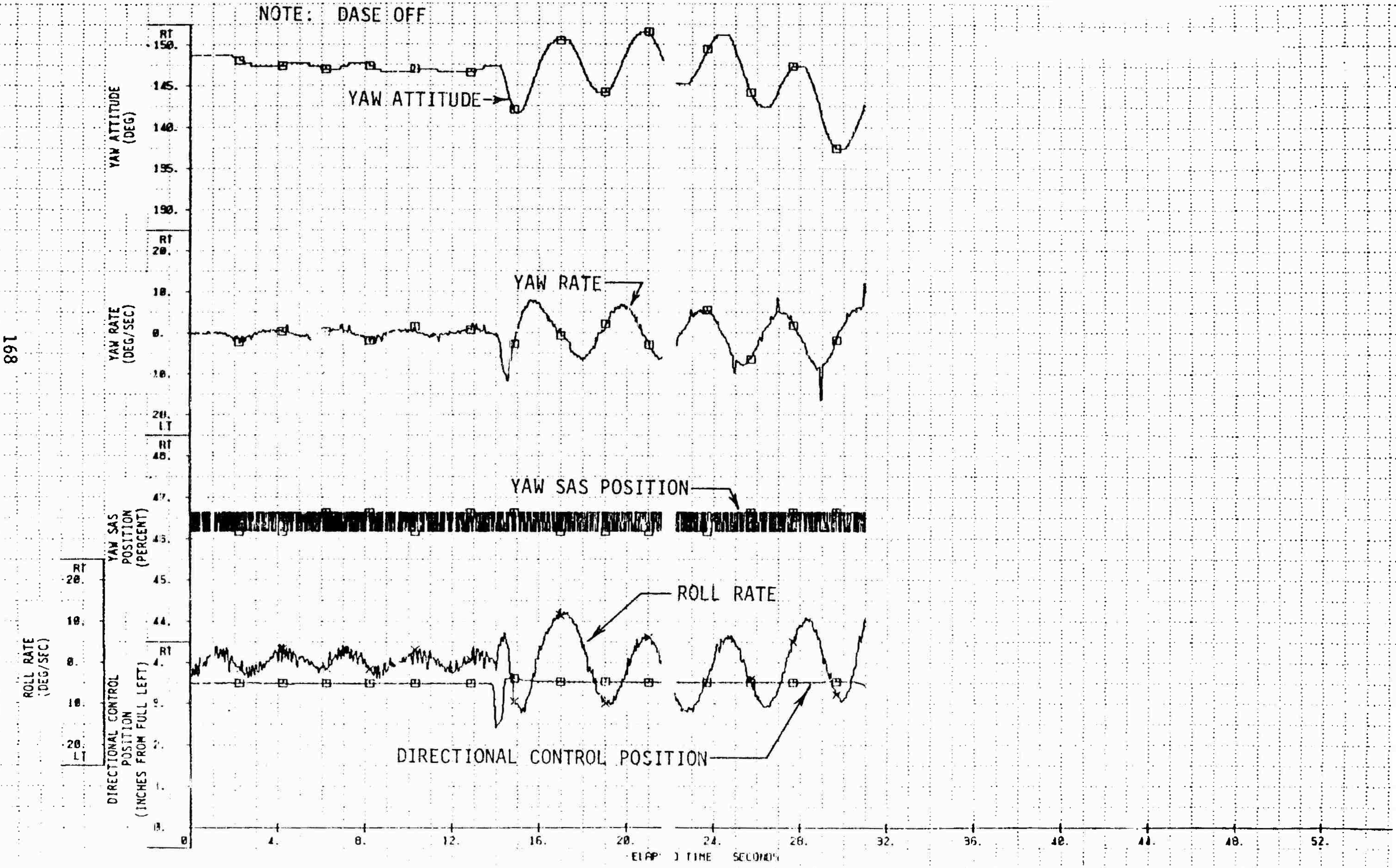


FIGURE 41A
RECOVERY FROM AUTOROTATION
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
15090	206.5 (AFT)	7180	22.0

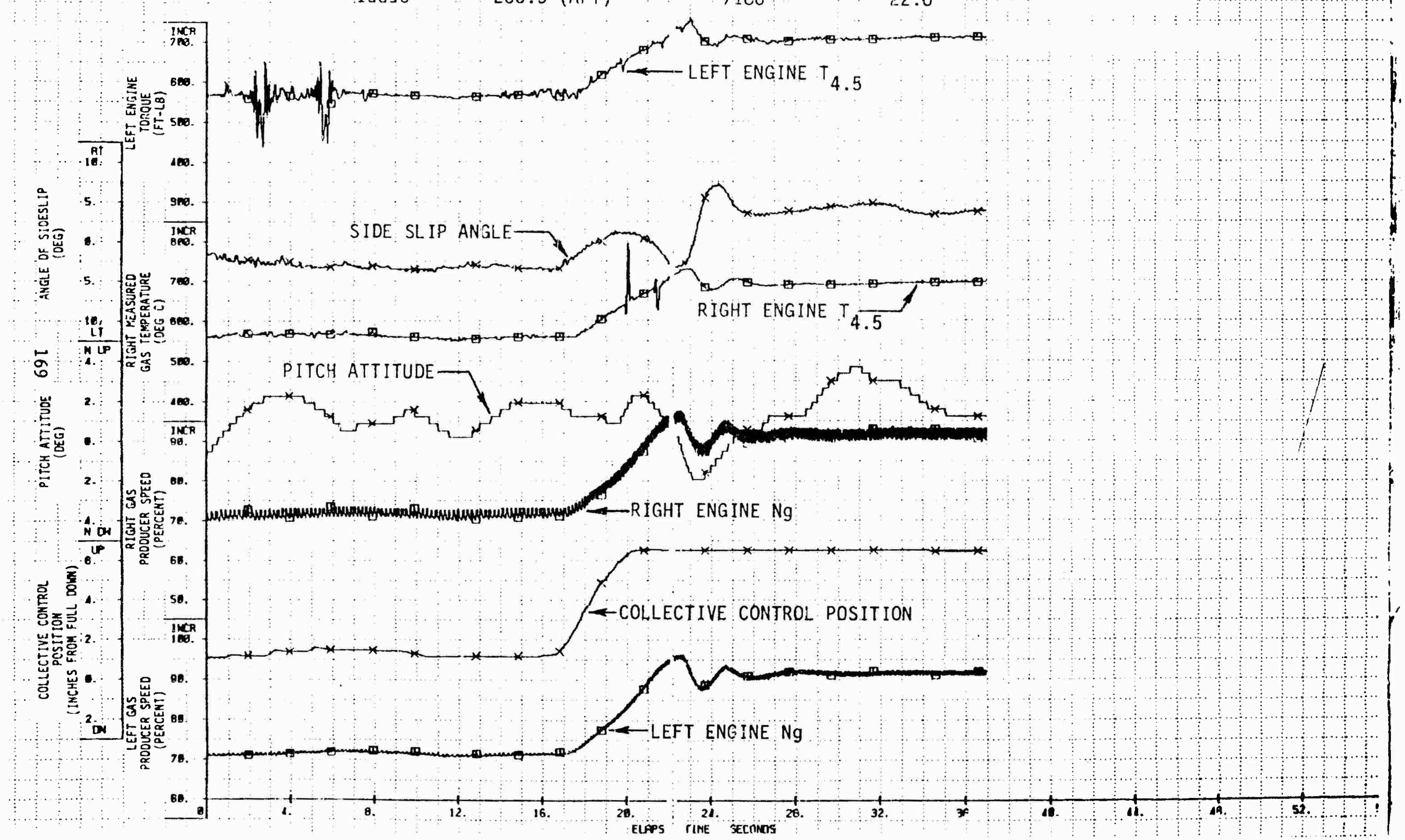


FIGURE 41B
RECOVERY FROM AUTOROTATION
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
15090	206.5 (AFT)	7180	22.0

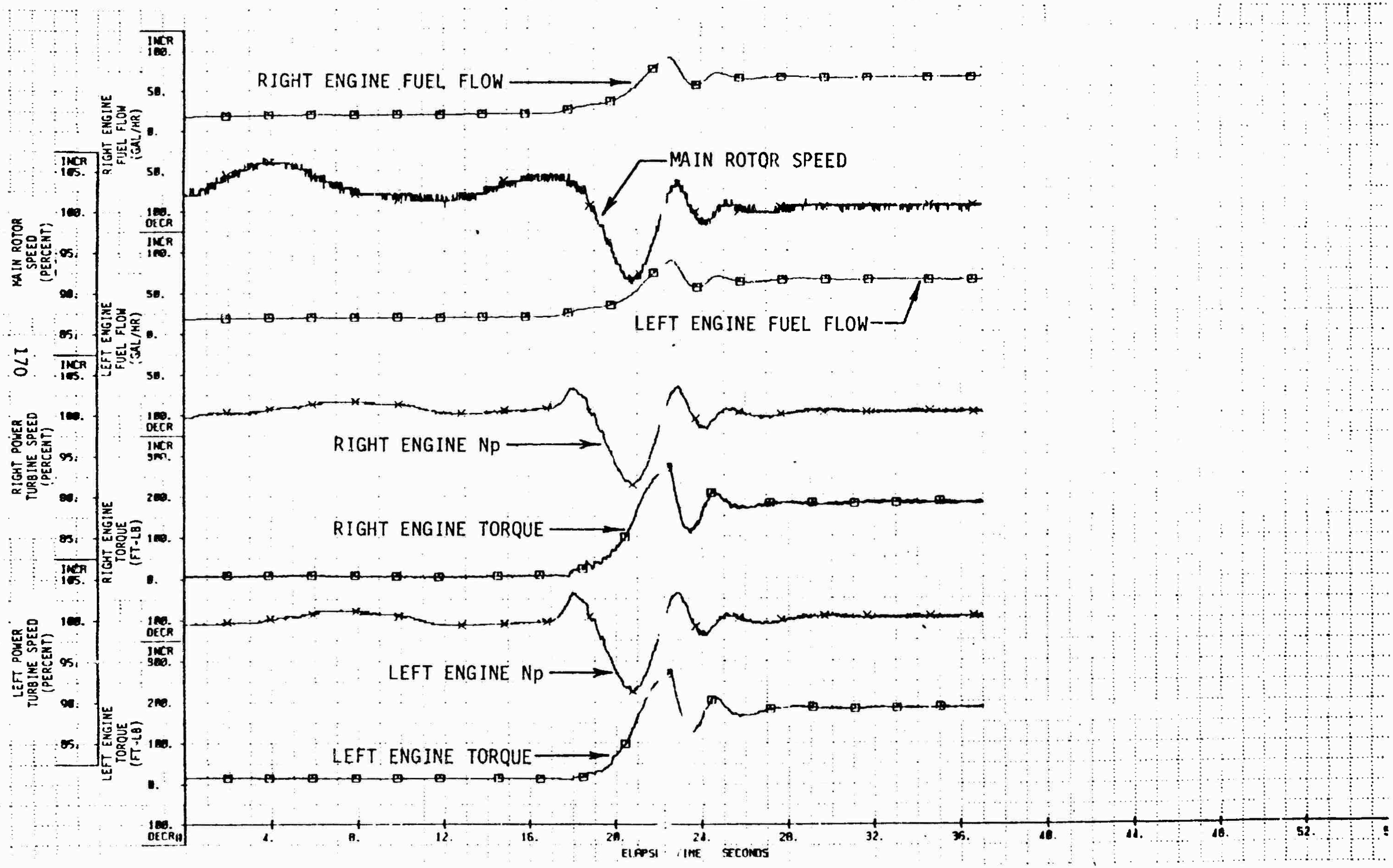


FIGURE 42
QUICK STOP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14800	206.1 (AFT)	7150	22.5

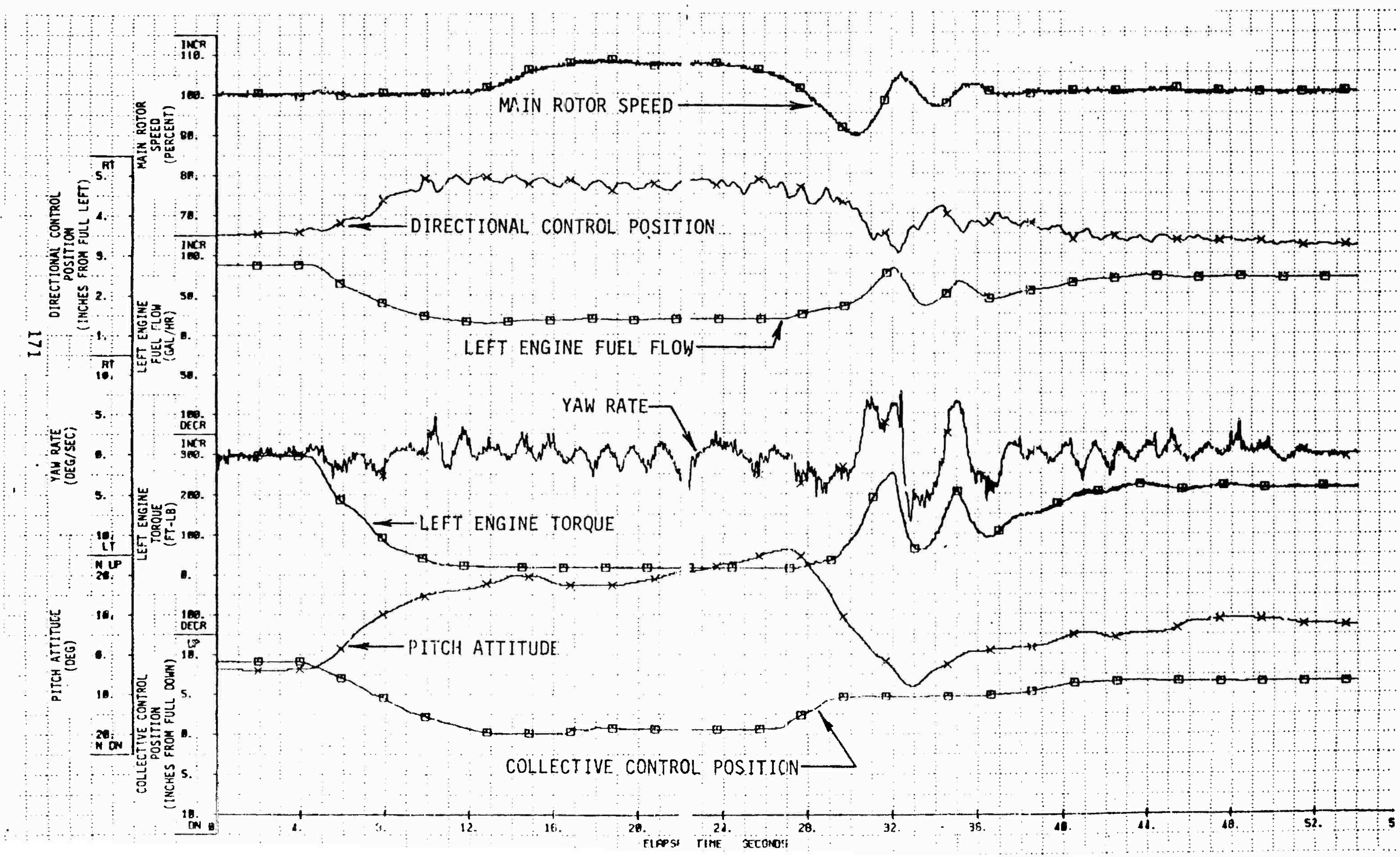


FIGURE 43A
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	7290	29.0

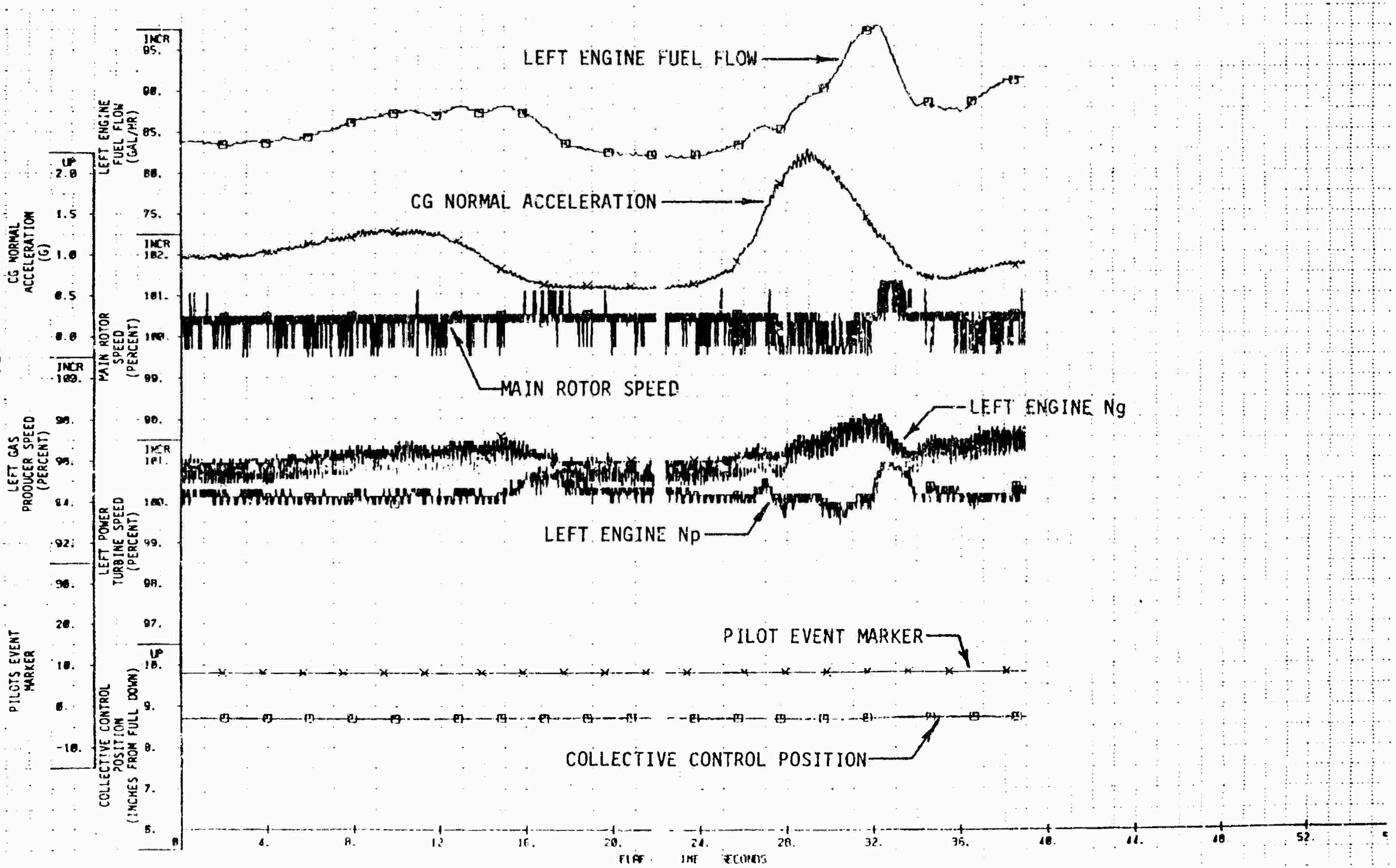


FIGURE 43B
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	7290	29.0

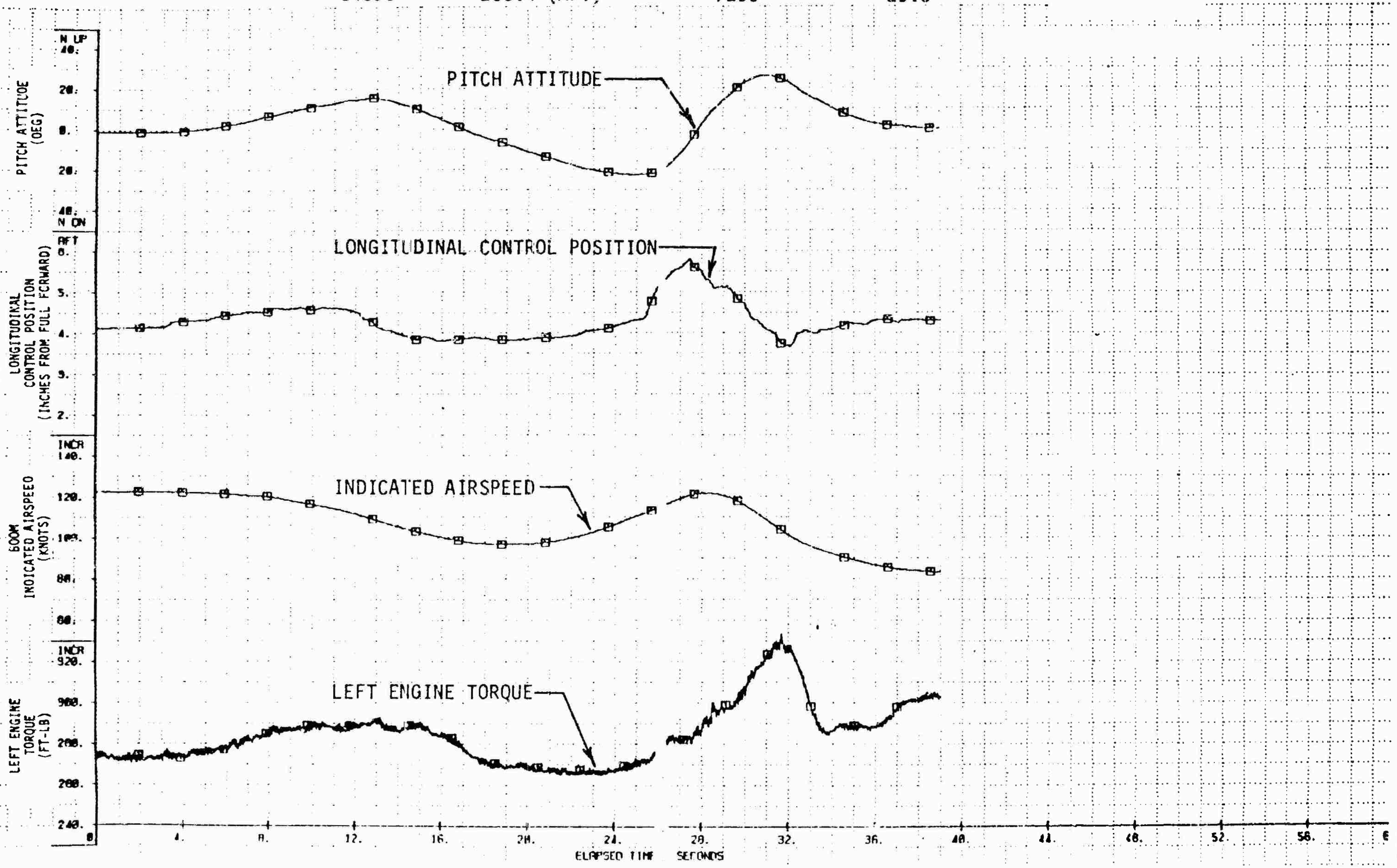


FIGURE 44A
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	6150	30.0

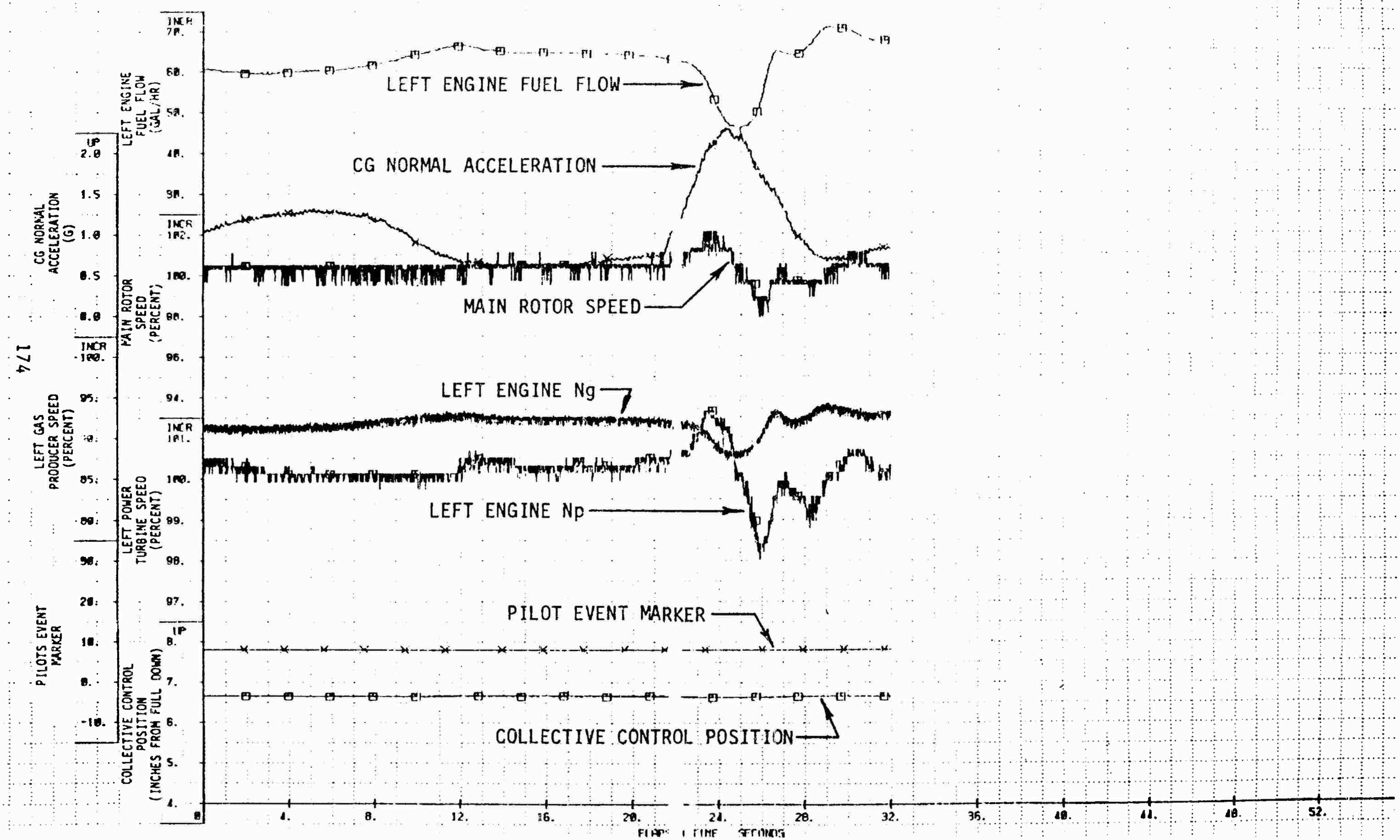


FIGURE 44B
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	6150	30.0

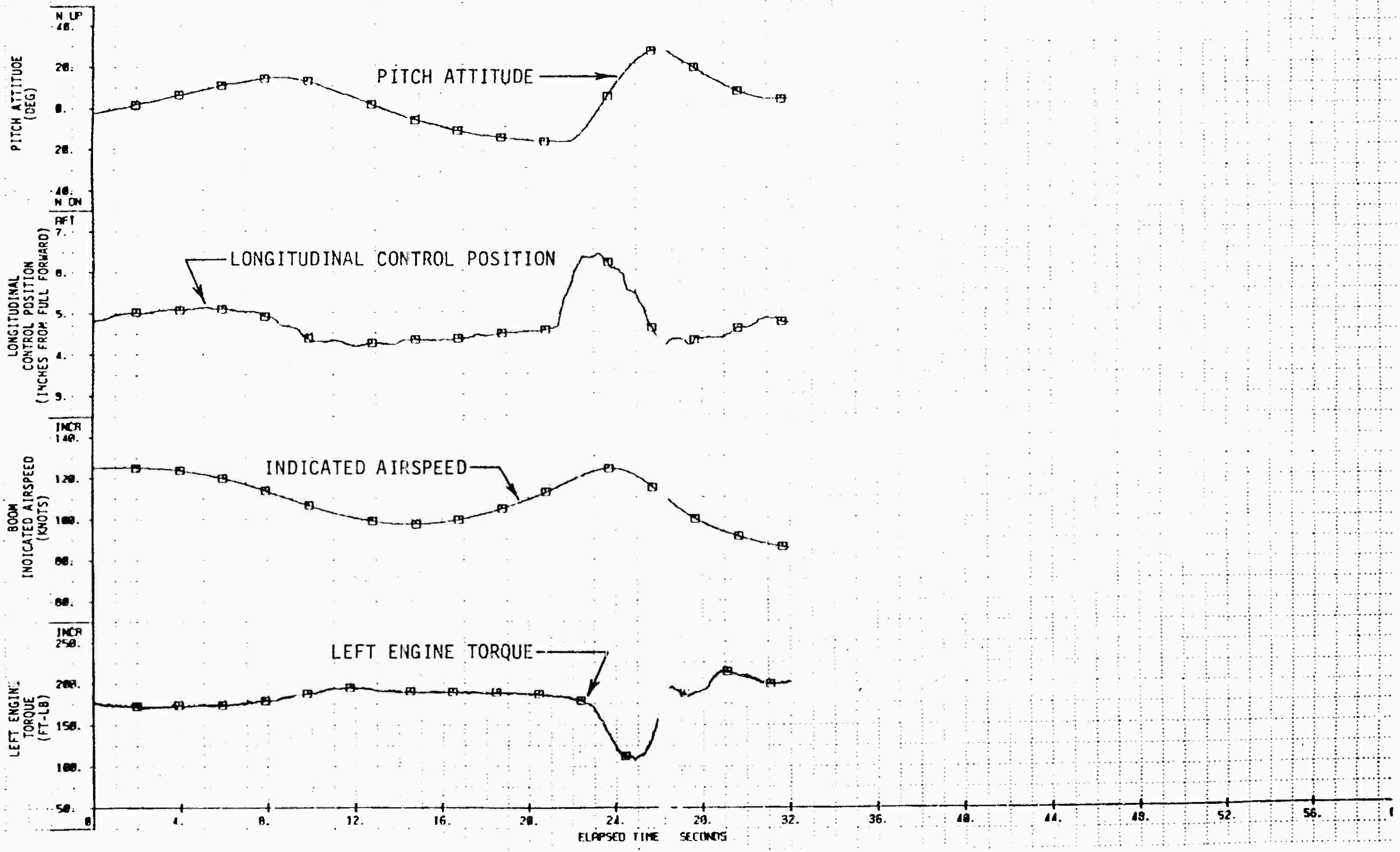


FIGURE 45A
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	6500	30.0

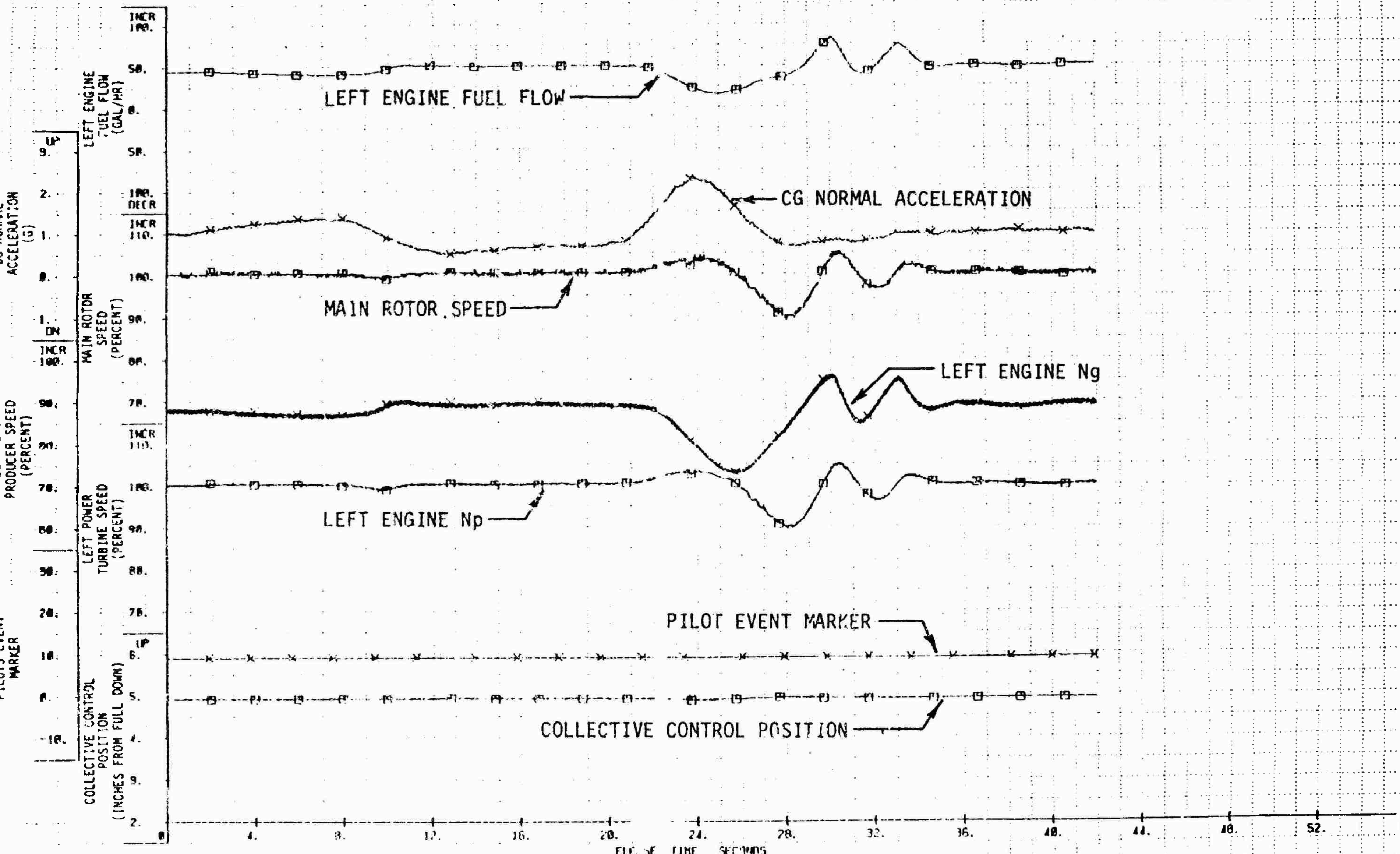


FIGURE 45B
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	6500	30.0

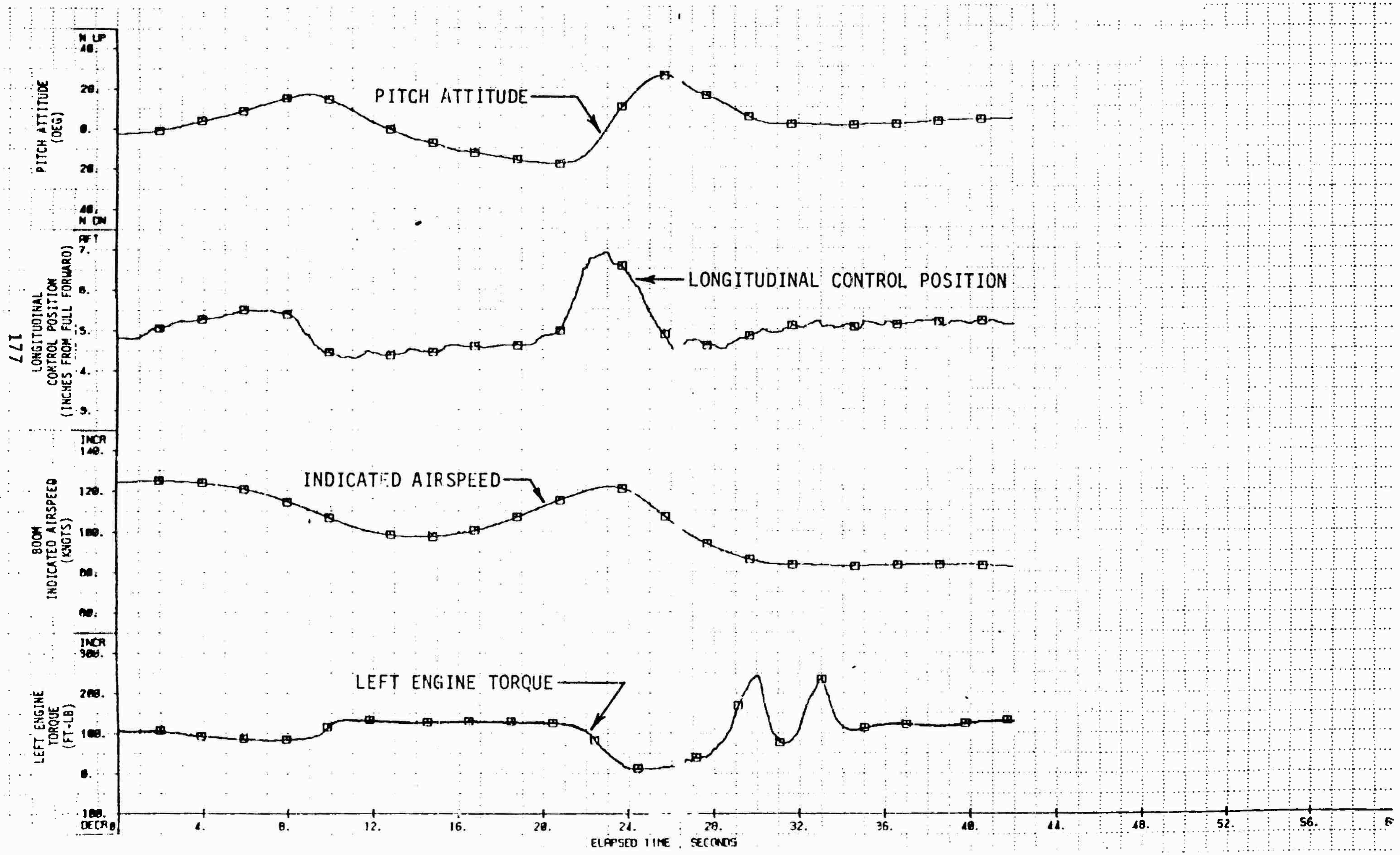


FIGURE 45C
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	5500	30.0

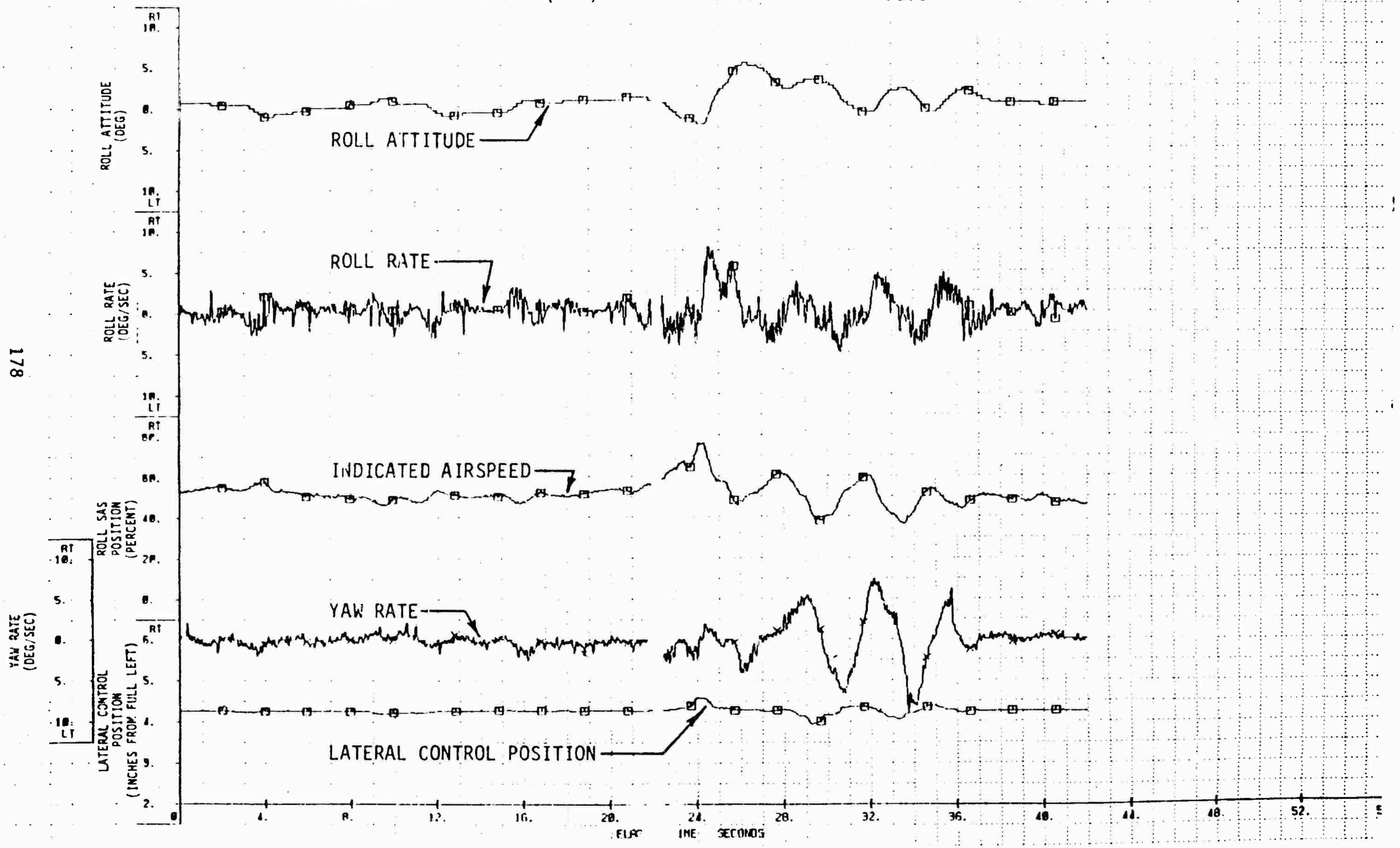


FIGURE 45D
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14690	206.4 (AFT)	6500	30.0

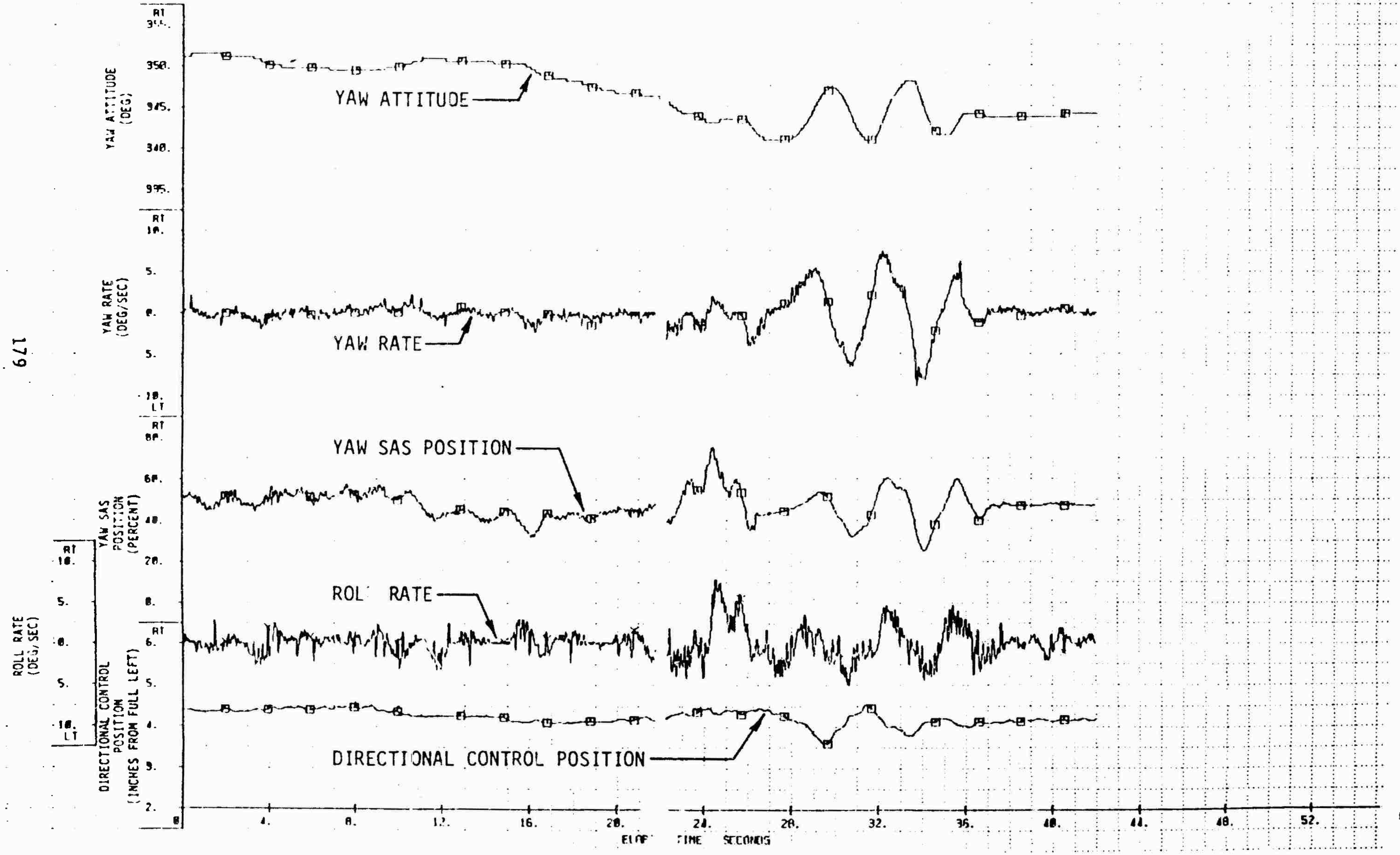


FIGURE 46A
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
15050	206.7 (AFT)	7030	22.0

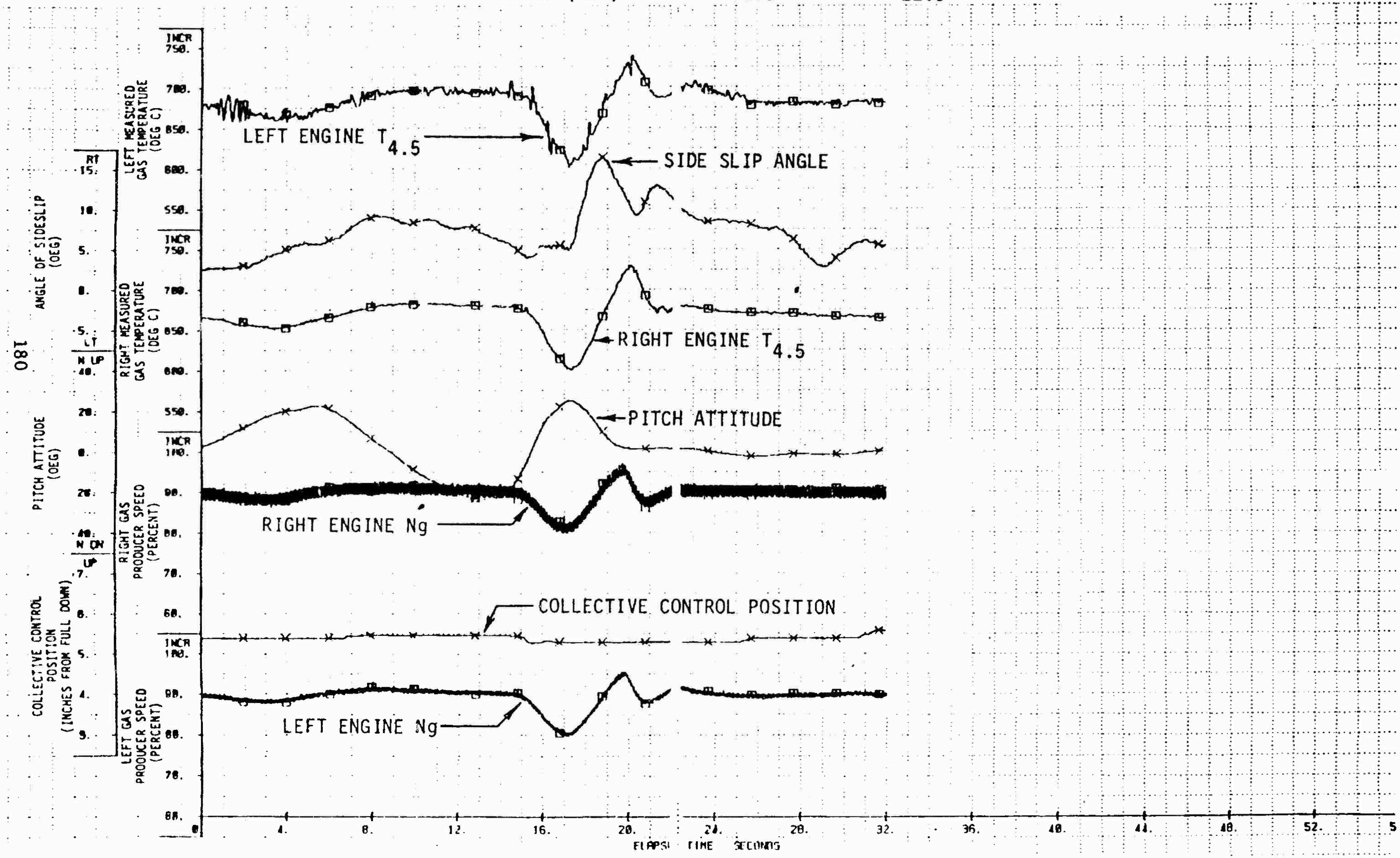


FIGURE 46B
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
15090	206.5 (AFT)	7030	22.0

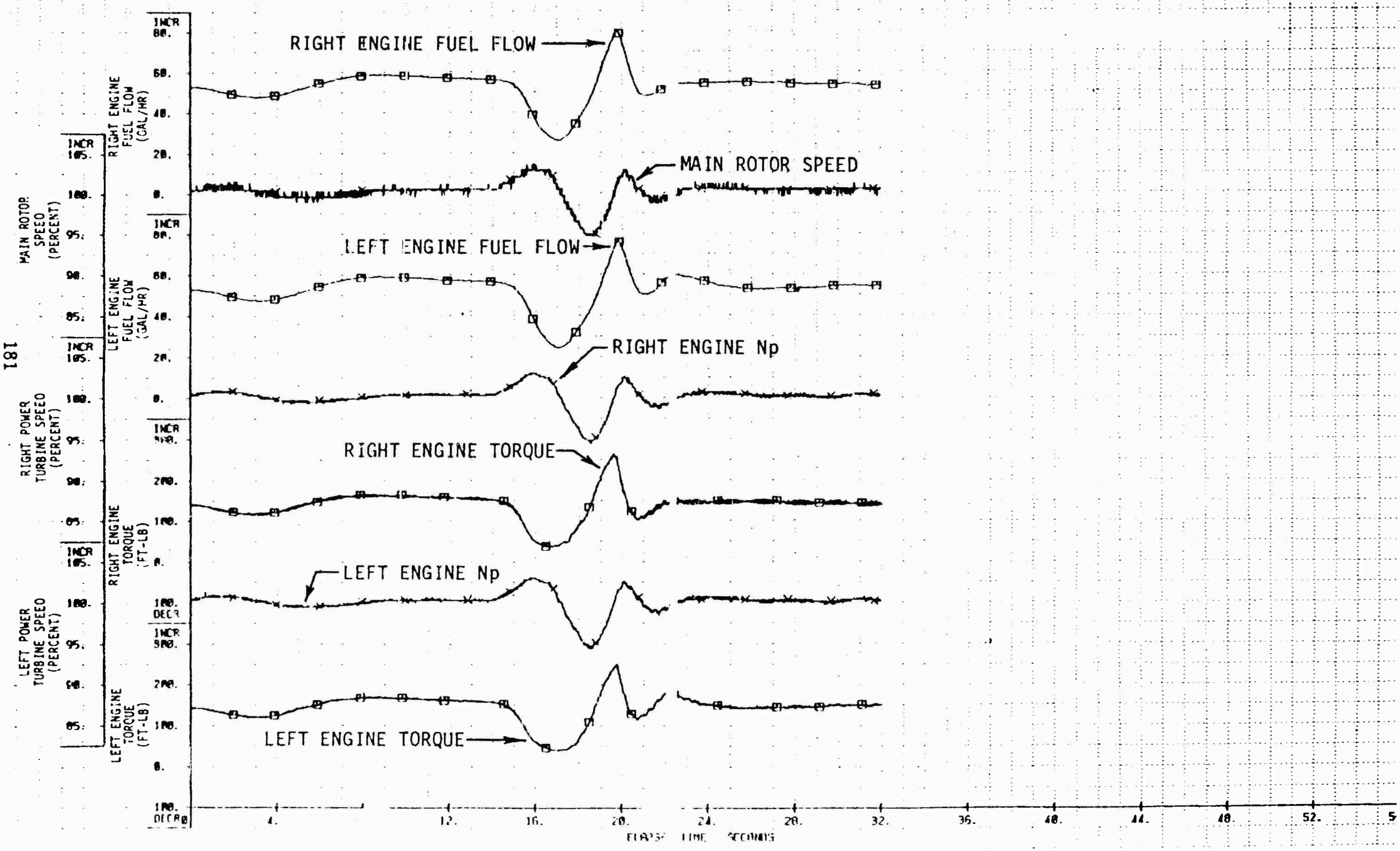


FIGURE 47A
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14770	206.4 (AFT)	6460	25.0

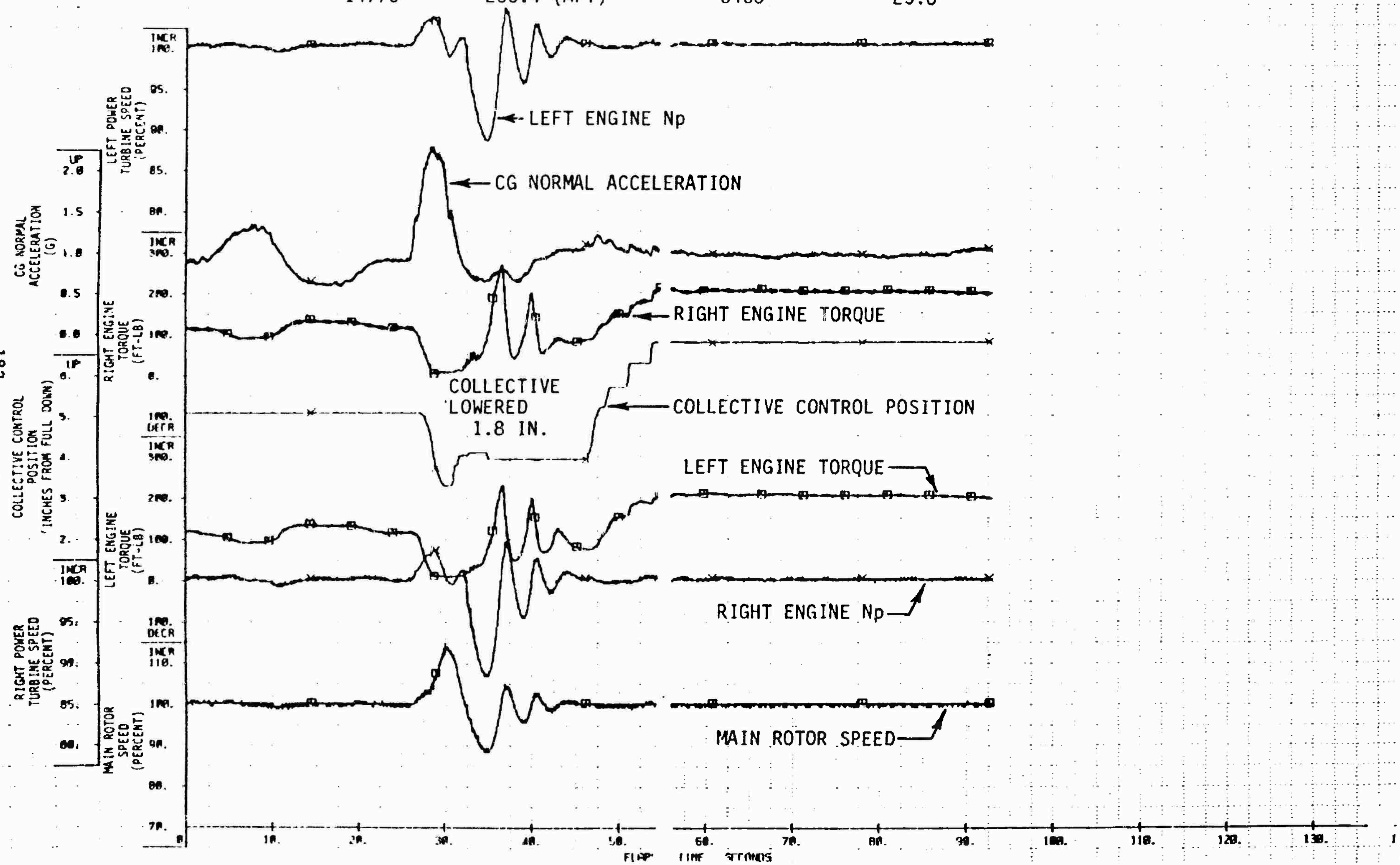


FIGURE 47B
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14770	206.4 (AFT)	6460	25.0

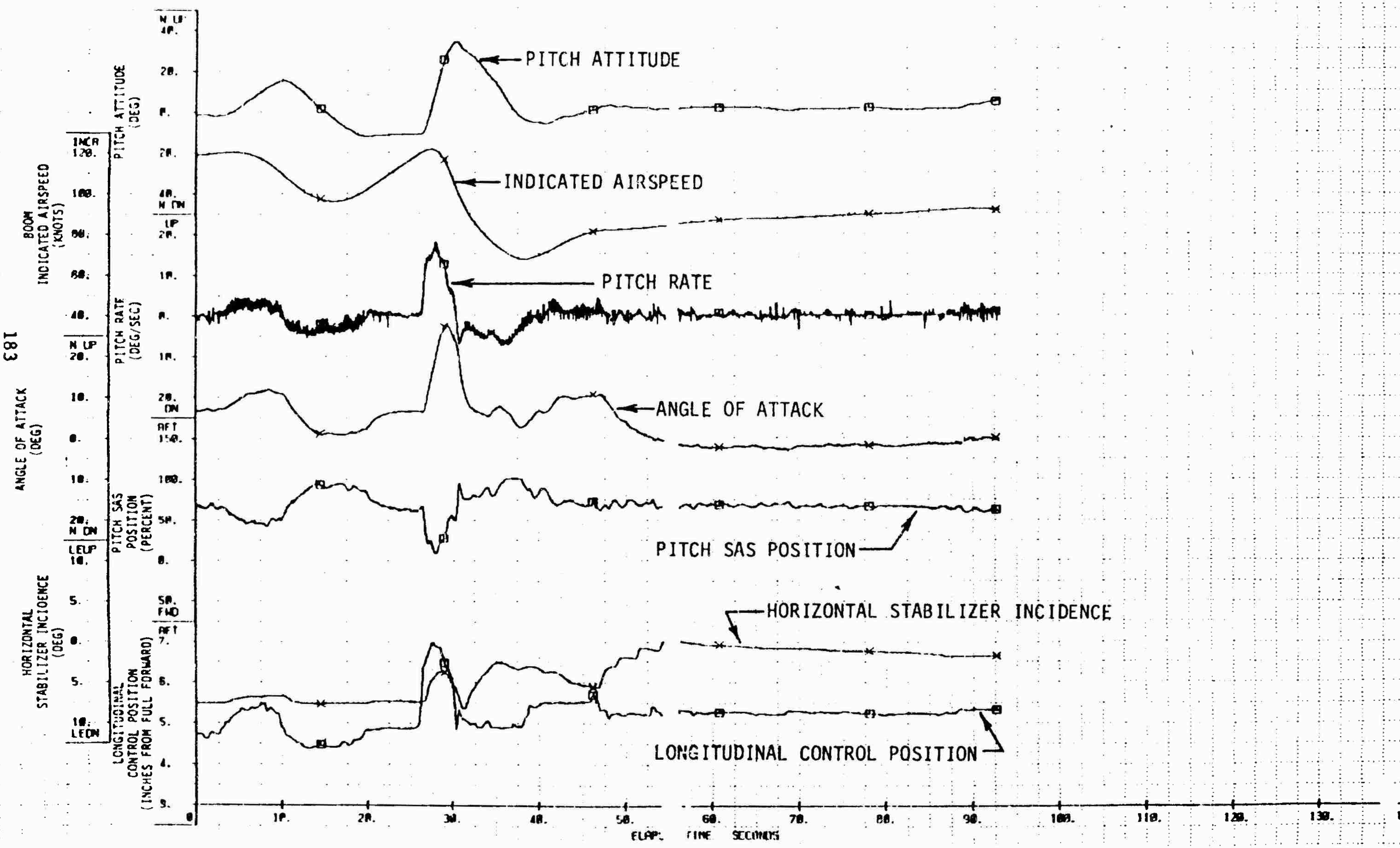


FIGURE 48
LEFT TURN
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)
14800	206.1 (AFT)	4380	28.0

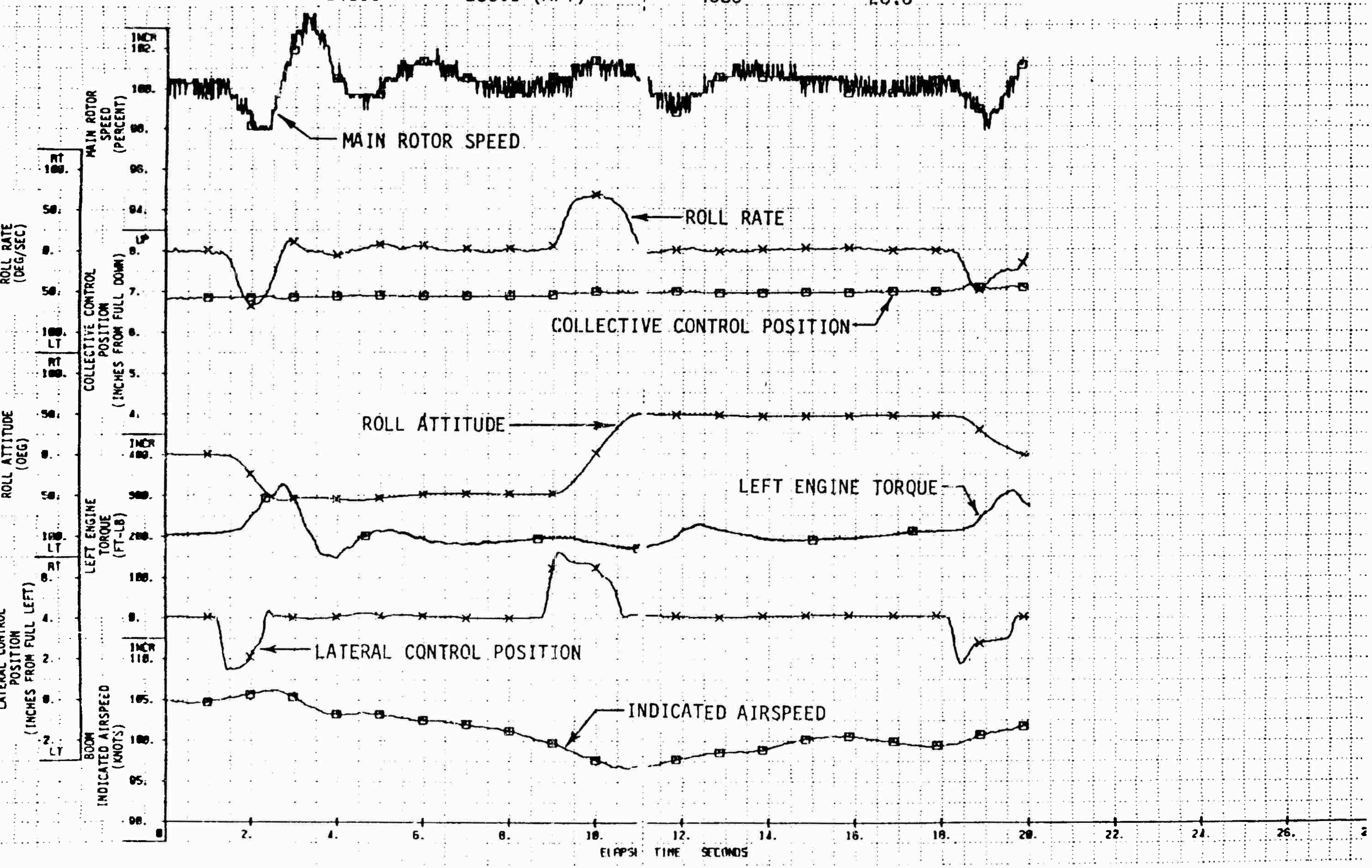


FIGURE 49
SIMULATED ENGINE FAILURE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)	LONG. CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM FLIGHT CONDITION
14690	206.4 (AFT)	7460	29.0	LEVEL

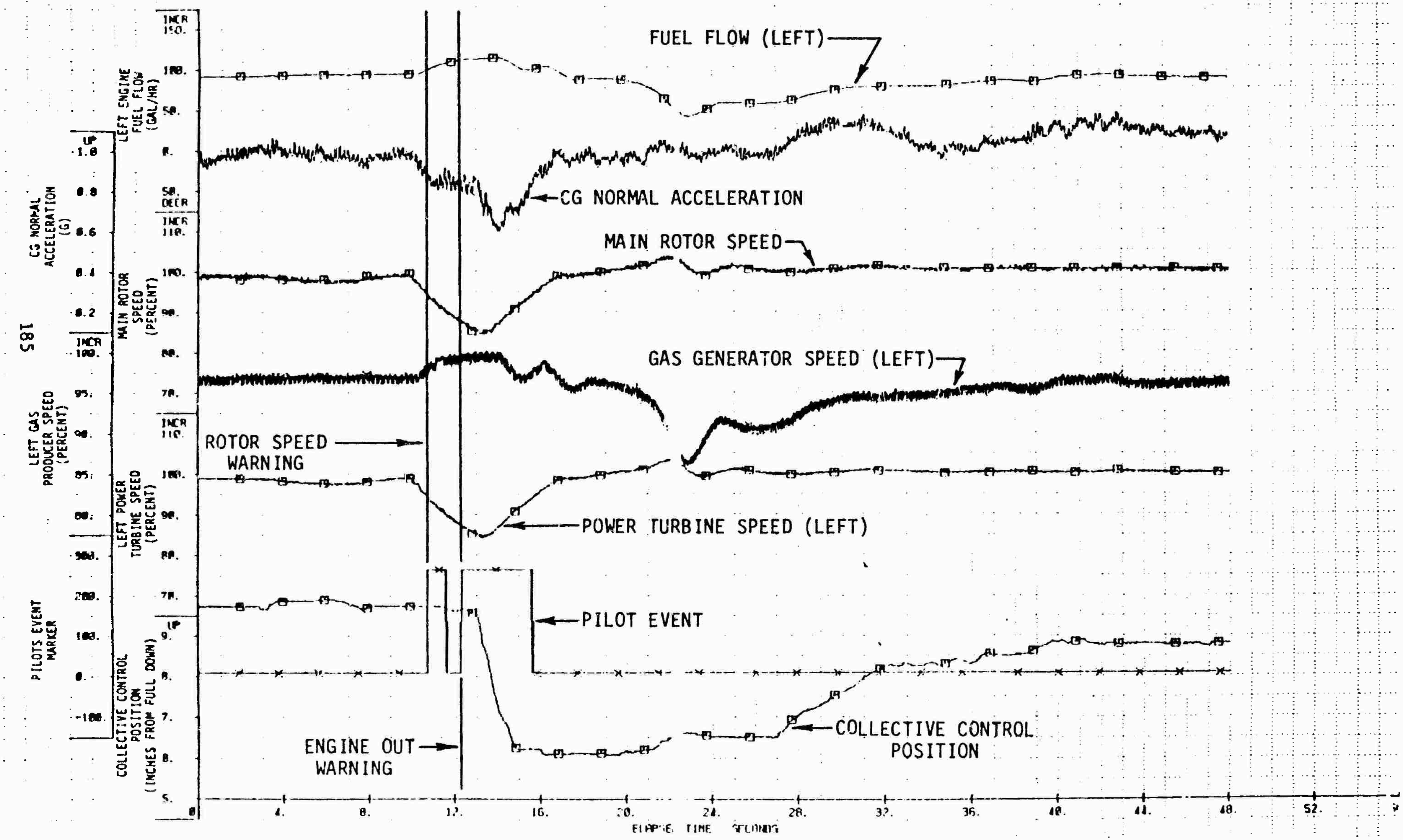


FIGURE 50
VIBRATION CHARACTERISTICS
AH-64A USA S/N 82-23355
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14770	208.7 (AFT)	10000	13.5	290	LEVEL

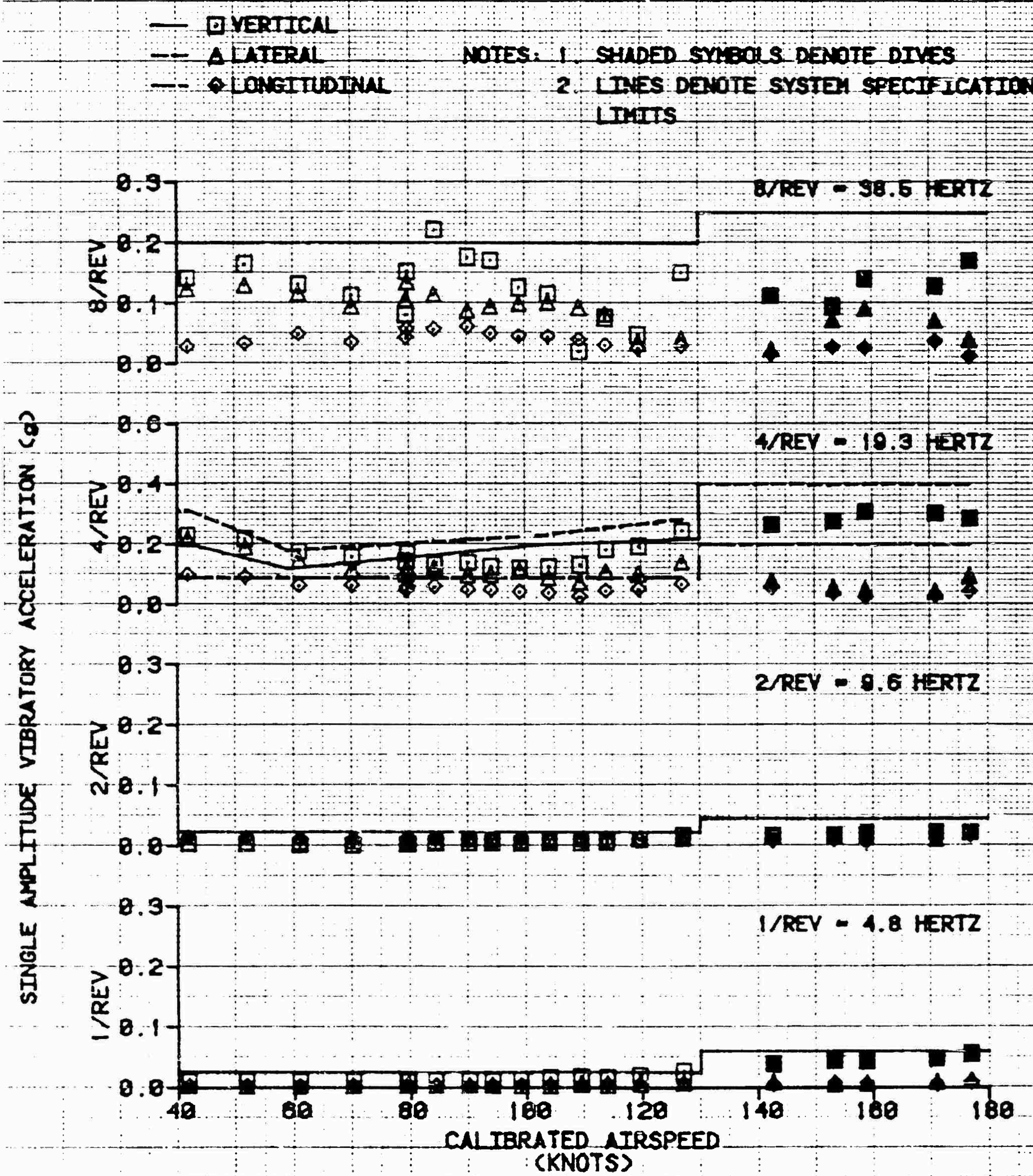


FIGURE 51
VIBRATION CHARACTERISTICS
AH-64A USA S/N 82-23355
COPILOT SEAT

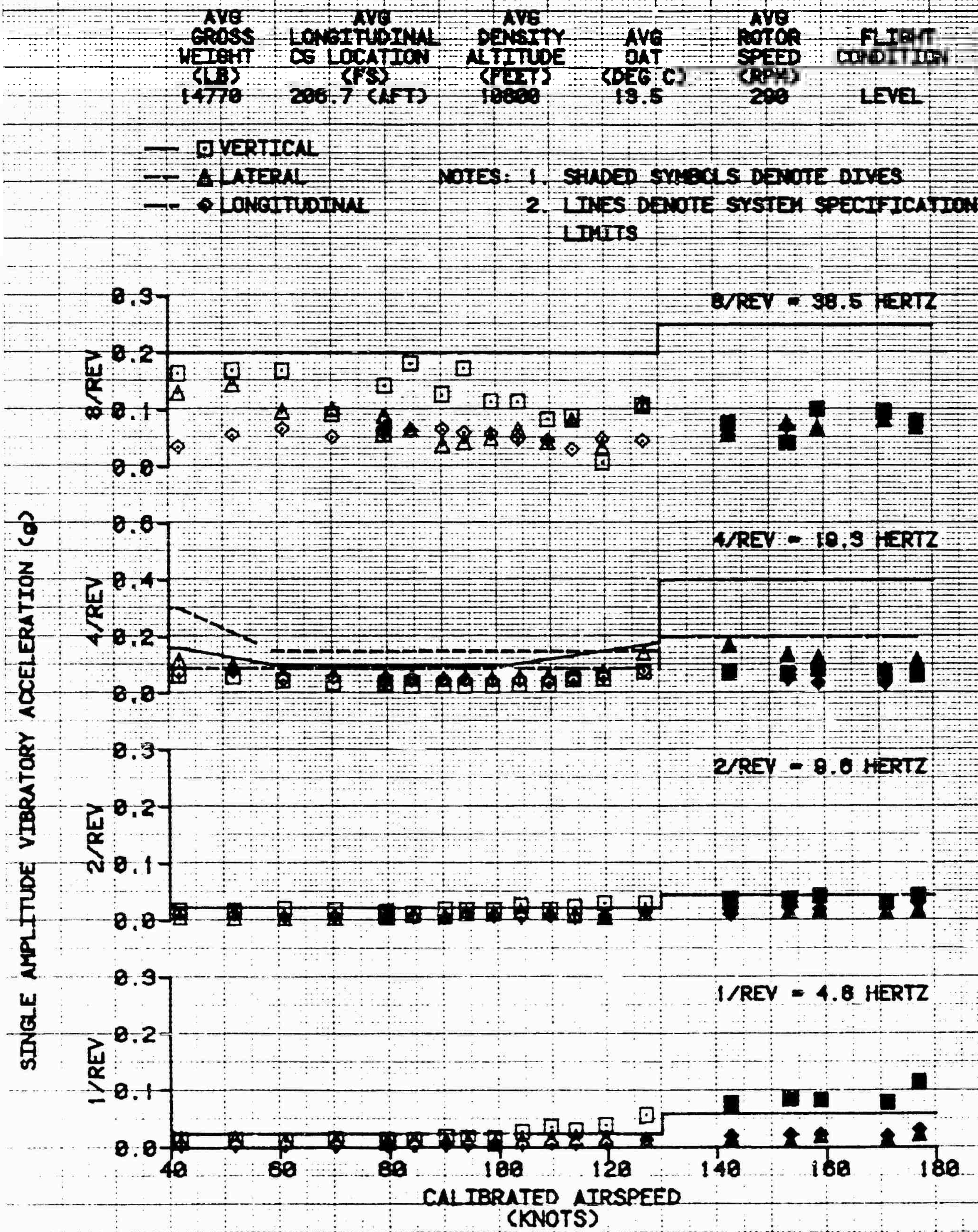


FIGURE 52
VIBRATION CHARACTERISTICS
AH-64A USA S/N 82-23355
AIRCRAFT CENTER OF GRAVITY

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14770	206.7 (AFT)	10000	13.5	290	LEVEL

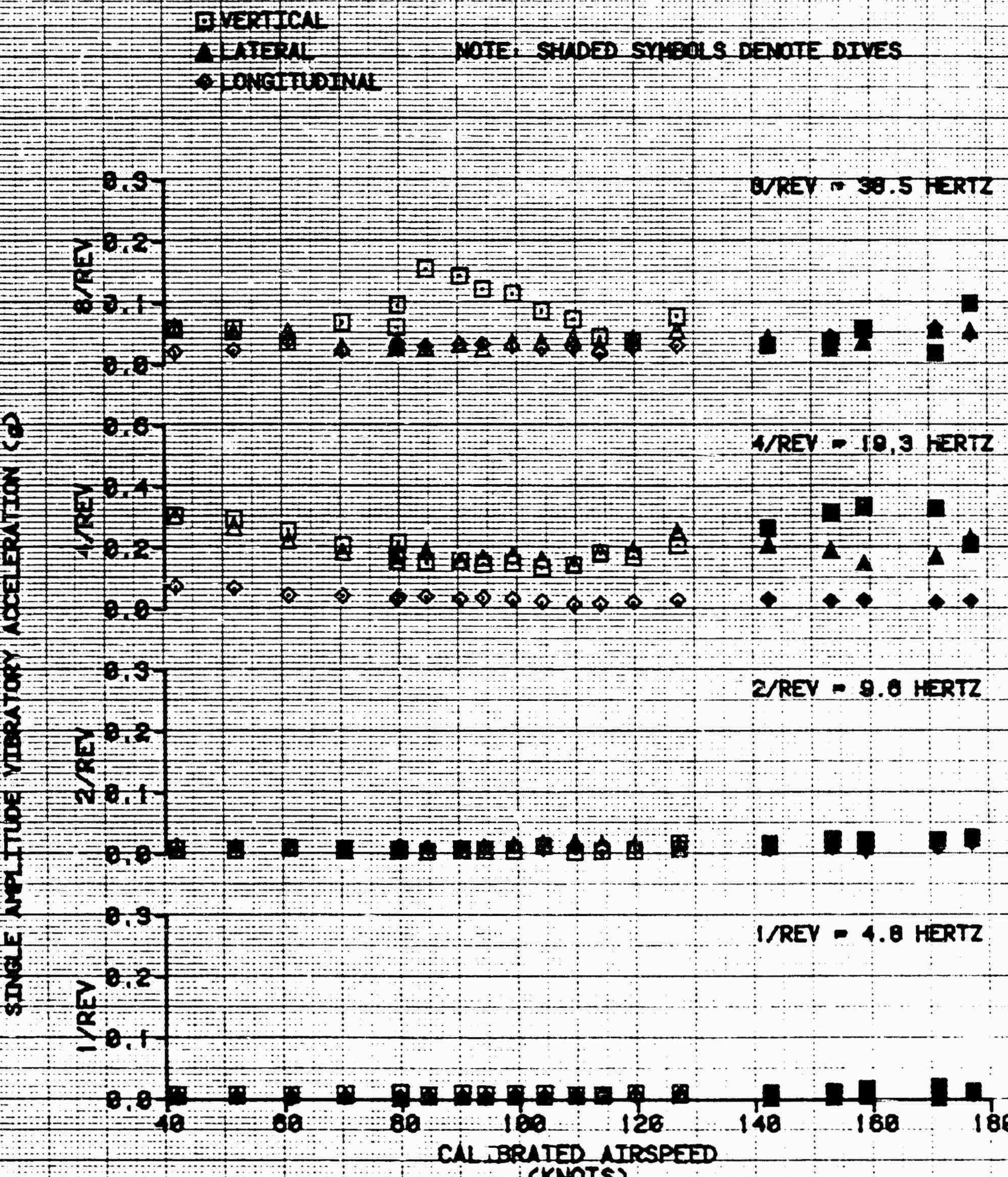


FIGURE 53
AIRSPEED CALIBRATION
AH-64A USA S/N 82-23355
PILOT PITOT-STATIC SYSTEM

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14880	208.4 (AFT)	7250	20.5	290	LEVEL

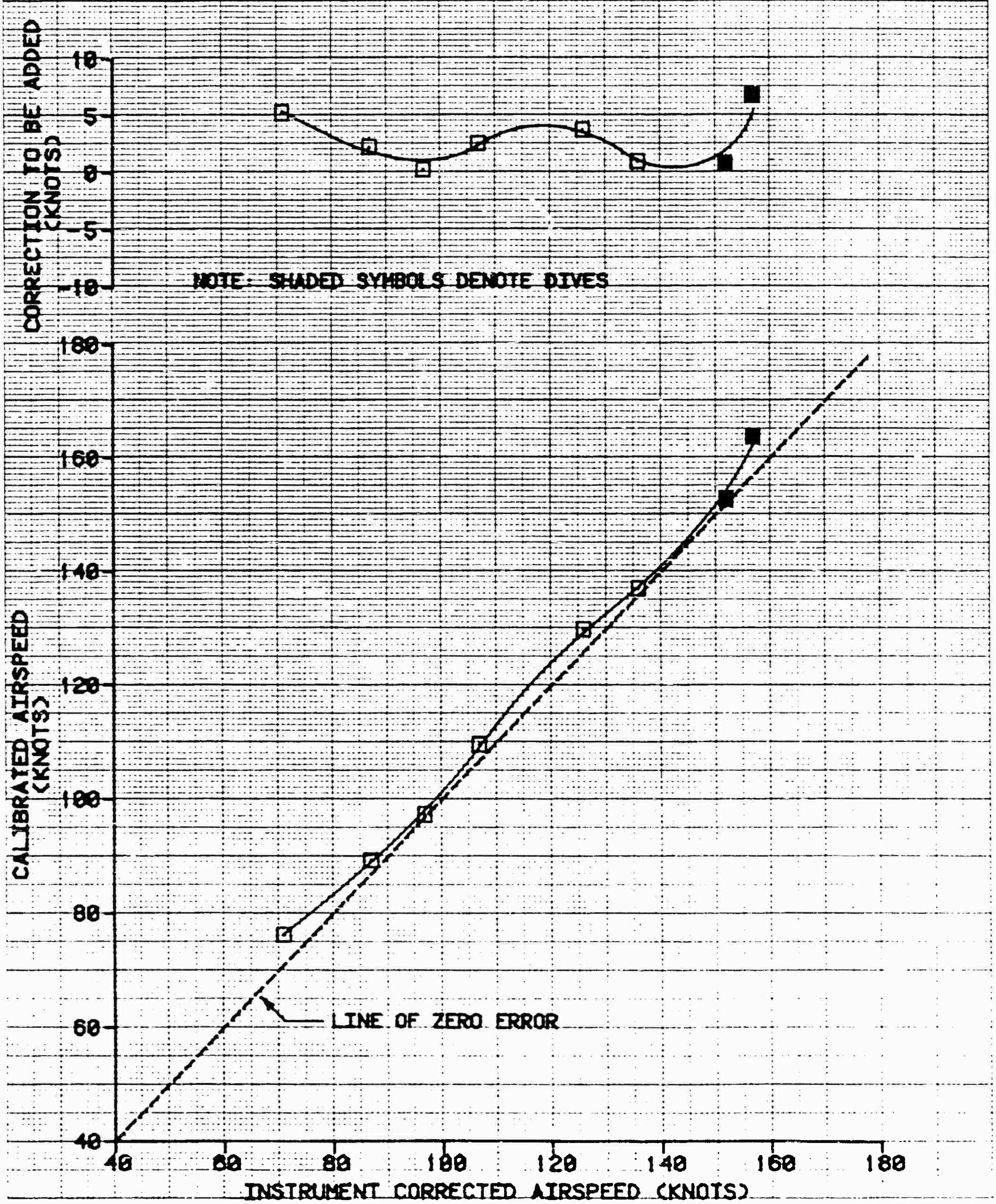
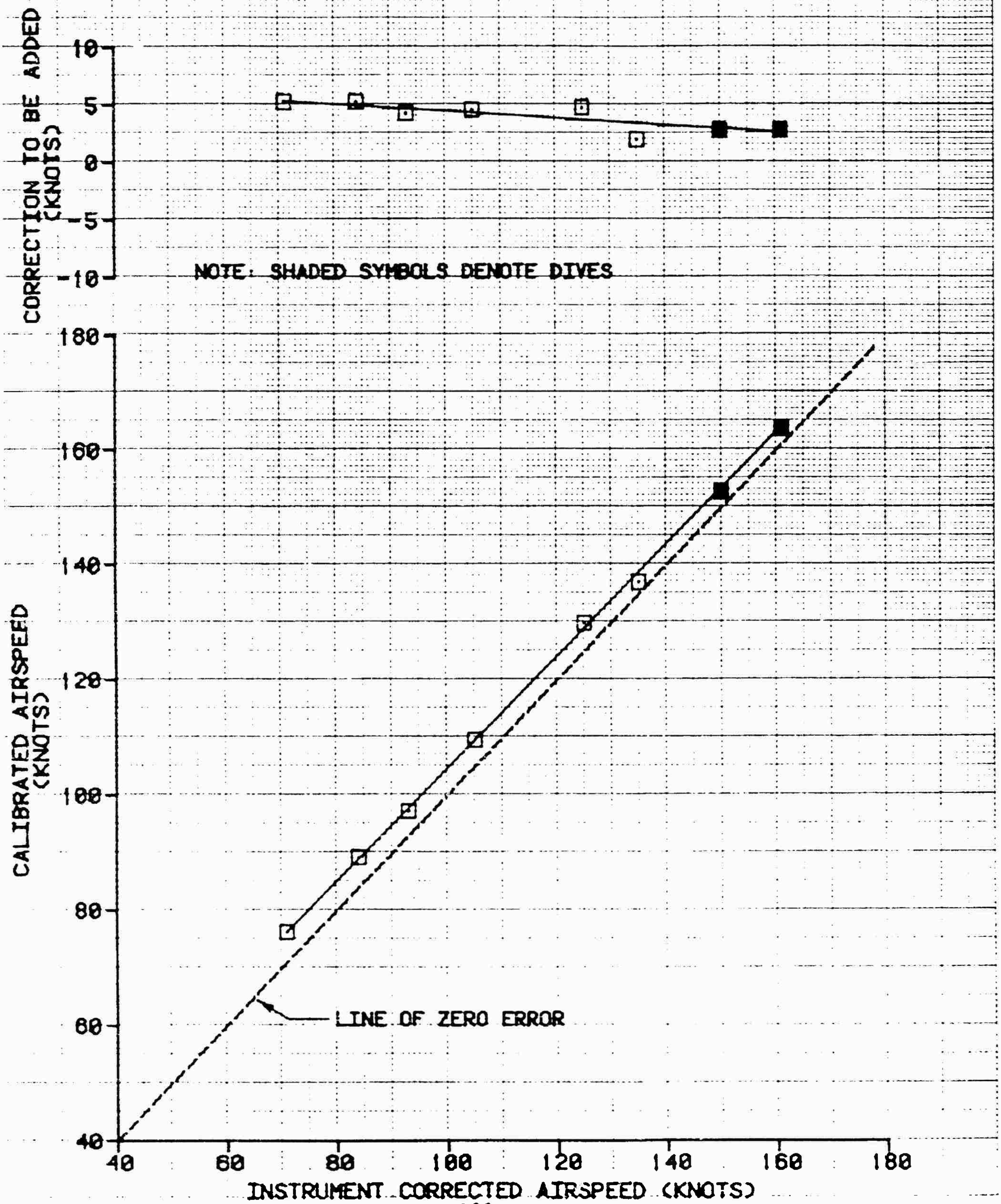


FIGURE 54
AIRSPEED CALIBRATION
AH-64A USA S/N 82-23355
COPILOT PITOT-STATIC SYSTEM

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14880	206.4 (AFT)	7250	20.5	290	LEVEL



APPENDIX F. EQUIPMENT PERFORMANCE REPORTS

NUMBER	SUBJECT
84-10-1	SDC Failure
84-10-2	Primary hydraulic pump failure
84-10-3	Engine anti-ice valve failure
84-10-4	SDC failure
84-10-5	Cracks in vertical stabilizer support mounts
84-10-6	RPM warning box
84-10-7	Aft fuel tank contamination
84-10-8	Heading attitude reference system
84-10-9	DASE computer FD/LS check failures
84-10-10	IGB grease expulsion
84-10-11	Cracks in rubber of #2 blade damper
84-10-12	Lack of pitch reference in VDU
84-10-13	EMI in caution panel
84-10-14	Intermittant failure of the tail wheel to unlock
84-10-15	Failure of the HARS to consistently align with correct magnetic heading
84-10-16	The failure of the ADF set to provide aural identification and accurate bearing information
84-10-17	Failure of the #1 nose gear box

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