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AVSCOM PROJECT NOS. 70-02 and 71-17
USAASTA PROJECT NOS. 70-02 and 71-17

ARMY PRELIMINARY EVALUATION I
AND
RESEARCH AND DEVELOPMENT ACCEPTANCE TEST I
AH-56A CHEYENNE COMPOUND HELICOPTER

FINAL REPORT

FLOYD L. DOMINICK JR.
PROJECT ENGINEER

GARY L. BENDER
PROJECT ENGINEER

RANDY D. McCLELLAN
ENGINEER

GARY C. HALL
LTC, TC
US ARMY
PROJECT OFFICER/PILOT

PAUL G. STRINGER
MAJ, CE
US ARMY
PROJECT PILOT

MARVIN W. BUSS
PROJECT PILOT

MARCH 1972

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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AVSCOM PROJECT NOS. 70-02 and 71-17
USAASTA PROJECT NOS. 70-02 and 71-17

(14) 70-02, USAASTA-71-17

(6)

ARMY PRELIMINARY EVALUATION I AND RESEARCH AND DEVELOPMENT ACCEPTANCE TEST I, AH-56A CHEYENNE COMPOUND HELICOPTER.

(9) FINAL REP[redacted]T. 30 Jan - 23 Dec 71,

(10)

FLOYD L. DOMINICK JR.,
[redacted]

GARY L. BENDER,
[redacted]

RANDY D. McCLELLAN,
[redacted]

GARY C. HALL
[redacted]

PAUL G. STRINGER
[redacted]

(11) MAR[redacted]72

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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FOREWORD

During the testing of the AH-56A compound helicopter at Yuma Proving Ground, Arizona, the aircraft and special instrumentation were maintained by Lockheed-California Company personnel. Structural engineering support was provided by the Flight Standards and Qualification Directorate, US Army Aviation Systems Command.

ABSTRACT

The Army Preliminary Evaluation I and a portion of the Research and Development Acceptance Test I were conducted periodically on the AH-56A Cheyenne compound helicopter by the US Army Aviation Systems Test Activity between 30 January and 23 December 1971. These engineering tests were divided into five distinct phases to permit Army evaluation of the aircraft at various stages of the contractor development program. Primary test objectives were to gather stability and control data to provide an early assessment of the AH-56A, to assist in determining flight envelopes for future Army tests, and to examine previously identified problem areas. None of the rotor dynamic instabilities previously encountered in the contractor's development program were noted during these tests. Lateral control migration with airspeed was not objectionable. The capability of the pusher propeller to provide rapid deceleration and to control airspeed independently of dive angle is an excellent feature. Five deficiencies and 54 shortcomings were identified. The deficiencies are (1) excessive pilot workload due to unacceptable static lateral-directional stability characteristics at low airspeed seriously impairs the capability to operate at minimum altitudes unaffected by conditions of darkness or adverse weather, (2) uncommanded aircraft motion and loss of control during some maneuvering flight conditions, (3) rapid rate of rotor speed decay following simulated engine failures which allows the rotor speed to drop below the present transient limit, (4) inadequate directional control margins in sideward flight, and (5) excessive vibration levels in portions of the flight envelope. Correction of the deficiencies should be a prerequisite for an airworthiness release for operational Army aviators, and correction of the shortcomings is desirable. Two deficiencies warrant a reduction of the flight envelope size for future Army tests until correction of those deficiencies is accomplished. Further testing of the AH-56A is recommended.

DEPARTMENT OF THE ARMY
HEADQUARTERS, US ARMY AVIATION SYSTEMS COMMAND
PO BOX 209, ST. LOUIS, MO 63166

2 3 JUN 1972

AMSAV-EFT

SUBJECT: Army Preliminary Evaluation I and Research and Development Acceptance Test I, AH-56 Cheyenne Compound Helicopter

Project Manager
Advanced Attack Helicopter
PO Box 209
St. Louis, Missouri 63166

1. The purpose of this letter is to transmit with the subject report, AVSCOM comments on the deficiencies, shortcomings and recommendations presented by the test agency as the result of their airworthiness testing of the AH-56. The developmental nature of the test vehicle must be kept in mind, however, successful completion of contractor and Army test efforts are intended to substantiate airworthiness of the vehicle for production consideration except where configuration changes of the production design preclude substantiation. Additionally, it should be noted that due to funding and schedule restraints on the restructured RD&E contract, the amount of airworthiness testing conducted by the Army on the AH-56 to date is somewhat less than for normal programs.

2. Analysis of the deficiencies listed in paragraph 148 is as follows:

a. The uncommanded aircraft motion and occasional loss of control during maneuvering flight resulting from blade moment stall is the most serious deficiency. Even though safety may be enhanced by further restricting the flight envelope in terms of reduced maneuver load factors, correction of this problem is a valid prerequisite for airworthiness release to operational Army units. The upcoming program with the advanced mechanical control system is intended to provide acceptable solutions to this aircraft limitation.

b. Although important vibration improvements have been made throughout the AH-56 development program, compliance with specification vibration requirements remains essential to an adequate reliable weapon system with proper man-machine interface.

AMSAV-EFT
SUBJECT: Army Preliminary Evaluation I and Research and Development Acceptance Test I, AH-56 Cheyenne Compound Helicopter

c. The severity of the poor static lateral-directional stability below 100 knots is dependent on the amount of nap-of-the-earth operation intended under night or adverse weather conditions. Since this problem is the culmination of several stability weaknesses (directional stability, dihedral effect, and side-force characteristics) improvement in any one could provide an airworthy vehicle for such low speed, poor visibility operation with acceptable pilot workload.

d. The incorporation of the automatic propeller pitch reduction with low rotor speed provides a tool for potential elimination of the high rotor speed decay deficiency thru proper optimization of the system. Modified pilot technique from normal helicopter operations is not unreasonable for a compound helicopter. Further understanding and test of the rotor system may permit a reduction in the minimum permissible RPM operation range. The current 15% operations limit is quite small compared to other Army helicopters. Therefore, several alternatives appear open to eliminate this deficiency.

e. The sideward flight control limitations appear to be the easier of the stated deficiencies to eliminate in that additional tail rotor blade angle would provide more control for the same pedal displacement. Evaluation of a 2-1/2 degree increase in travel is part of the Armed Helicopter Requirements Evaluation. This includes testing at the more critical high altitude condition. The safety of flight release for these tests did not include the recommended reduction in the maximum allowed speed per the subject report.

3. The shortcomings, specification nonconformance and recommendations, paragraphs 149, 150, 151 and 153 thru 167 respectively, are covered in the inclosure to this letter. We have commented in these areas only if we disagreed with the findings as stated in the report or felt additional commentary was necessary for emphasis. It is noted that the majority of the shortcomings identified fall into the category of handling qualities correctable thru control system optimization or reliability and maintainability.

4. The report will be released as is for publication and distribution on 6 July 1972 unless you desire to make any additions to it by footnotes or indorsement.

FOR THE COMMANDER:

John C Geary

1 Incl
as

JOHN C. GEARY
Colonel, GS
Director of Research,
Development and Engineering

AH-56A APE I and RDAT I

Review Comments

1. Paragraph 149 (Shortcomings Affecting Mission Accomplishment).

a. Subpara a thru f and o - The prime area for improvement as relates to these items is one of control system optimization and as such should be pursued in conjunction with the Advanced Mechanical Control System (AMCS) effort.

b. Subpara g - It is agreed that the static longitudinal stability is weak above 150 KCAS; however, it is not considered deserving of any special emphasis beyond that relating or resulting from the AMCS effort.

c. Subpara h thru j - It is agreed that the collective results of these items contribute to the objectionable lateral-directional characteristics; however, the weak side force characteristic in itself is not considered a problem of such significance as to require correction nor is the inherent sideslip considered excessive. It is of interest to note that the AH-1G exhibits a right inherent sideslip of 2 to 4 degrees in level ball centered flight (turn and bank indicator) and increased to 5 to 5.5 degrees for a true wings level (zero roll angle) attitude. Increased pilot workload is required to optimize rocket firing accuracy by firing at zero sideslip; however, rocket firing accuracy is believed to be adequate.

d. Subpara k and l - The specification indicates a preference for a pull force with increasing sideslip; however, it is not in fact a requirement. The longitudinal trim shift with a sideslip should not result in a pull force greater than 10 lb. or a push force greater than 3 lb. which is a MIL-F-8785 specification requirement.

e. Subpara m - Aircraft control by the pilot is essentially related to stick free maneuvering characteristics as opposed to stick fixed; therefore, the emphasis should be placed on improving the stick free characteristics during the AMCS effort (see paragraph 152.6).

f. Subpara n - Emphasis should be placed in minimizing the lift/roll coupling during the AMCS effort.

g. Subpara p - As a minimum, efforts should be initiated to provide to the pilot a warning of uncommanded aircraft response resulting from blade moment stall. If, for example, the boundary can be defined in terms of collective, normal load factor, and airspeed, then condition these to activate a warning device.

h. Subpara v and aj - This has also been a problem when accelerating from low speed flight. AVSCOM will investigate possible technique and equipment to monitor atmospheric and engine condition parameters to limit beta input and display power margin for the pilot.

i. Subpara x These were random instances where conditions were not repeated in any case. The apparent cause of this problem is the lack of crew station sealing and lack of ECU flow volume.

j. Subpara y thru af and ar thru bb - These are essentially reliability and maintainability problems, the correction of which can only be made possible at reduced cost if they are uncovered early in the airworthiness qualification program.

k. Subpara ag - Since the engine condition lever is not normally used in flight AVSCOM recommends that this control be installed as a throttle type lever adjacent to the left console.

l. Subpara ai - The major portion of this problem lies in the location of this control switch. The use of the alternate N_f control becomes most critical when gross power changes are made. This creates a requirement that the increase/decrease switch be located on the collective control head so that it can be operated when the pilot is using his collective and beta controls. The impact of using the current trim switch for this function and relocating the alternate N_f selector switch to the left console should be investigated. The latter mode will serve to prevent inadvertent operation of the system.

m. Subpara [illegible] - The pilot's annunciator panel contains the major portion of the caution lights with the exception of the fault locating panel located on the right rear bulkhead of the copilot/gunner's station. Because of the copilot/gunner's task load during a target tracking or firing station, the fault locating panel should be located in the pilot's station in direct view of the pilot. With the panel in this position, the pilot will have all caution lights within his visual envelope. If this change is incorporated, this feature will no longer be a problem.

n. Subpara ao - The restructured program did not provide for EMC/EMI testing as such. These types of tests would be required for complete airworthiness qualification to identify and correct such problems.

2. Paragraph 151 (Specification Nonconformance).

a. Subpara b - It is agreed that MIL-H-8501A may be interpreted such that the longitudinal control position and force stability with respect to speed did not meet the requirement; however, the specification does not indicate whether it is applicable to the local gradients or average gradient and the velocity range from trim is not set forth.

b. Subpara d - It is agreed that the sideslip angles required by MIL-H-8501A were outside the flight envelope released; however, within the range tested pedal position about zero sideslip was stable and essentially linear while the dihedral effect was essentially zero near 60 KCAS and became slightly positive at increased speeds. Future qualification should be directed toward a greater high speed sideslope envelope.

3. Recommendations - This Command is in general agreement with the recommendations set forth subject to the following additional comments:

a. Para 153 and 154 - Correction of deficiencies and shortcomings and comments thereto are provided under the appropriate paragraph.

b. Para 155 - Additional testing and analysis should be conducted to:

(1) Completely define the blade moment stall boundary and aircraft response to same throughout the flight envelope.

(2) Identify the best pilot technique for recovery from blade moment stall.

(3) Provide a warning to the pilot of impending uncommanded aircraft response due to blade moment stall.

c. Para 156 - The high speed maneuvering capability should be improved to meet the initial normal load factor requirement of 2.0g's at 150 KCAS per the QMR.

d. Para 163 - Testing sufficient to establish a sideward and rearward flight speed envelope for appropriate ranges of gross weight and density altitudes should be conducted.

e. Para 167 - A special design review as such of the tail boom is not warranted until a production design is available.

f. General Comment - The additional flight testing recommended is considered applicable only the the AMCS effort. During such testing, emphasis should be placed on blade moment stall characteristics, optimization of roll response and control characteristics, improved lateral-directional characteristics, improved high speed maneuvering capability, and autorotation/entry, descent, and landing characteristics.

DEPARTMENT OF THE ARMY
PROJECT MANAGER, ADVANCED ATTACK HELICOPTER, AMC
PO BOX 209, ST. LOUIS, MO 63166

AMCPM-AAH-TM-T 8 August 1972

SUBJECT: Indorsement to Army Preliminary Evaluation I and Research and Development Acceptance Test I, AH-56A Cheyenne Compound Helicopter Final Report, March 1972

Commanding General
US Army Aviation Systems Command

1. The following analysis pertains to the validity of the major findings, conclusions and recommendations of the Army Preliminary Evaluation I and Research and Development Acceptance Test I, AH-56A Cheyenne Compound Helicopter Report as submitted to Project Manager of the Advanced Attack Helicopter.

2. The objectives of these engineering tests of the AH-56A Cheyenne with the Improved Control System (ICS) were not intended to provide an airworthiness qualification for operational use. The primary objective has been to assess the flying qualities of the AH-56A in order to provide timely information required for successful completion of the development program. Further development of the AH-56A aircraft is scheduled under terms of Development Contract DAAE11-66-C-3667(H) before commitment to a production configuration is made.

3. The five (5) characteristics reported as deficient have been analyzed and the following conclusions reached regarding validity as they affect crew or aircraft safety, mission capability and planned testing:

a. STATIC LATERAL DIRECTIONAL STABILITY

(1) DEFICIENCY: Excessive pilot workload due to the unacceptable static lateral directional stability characteristics below 100 KIAS seriously impairs the capability to operate at minimum altitudes unaffected by conditions of darkness or adverse weather.

(2) ANALYSIS:

(a) Three static lateral directional characteristics (directional stability, dihedral effect, and sideforce) were reported individually as shortcomings with improvement desired to enhance the mission capability of the AH-56A. These characteristics have been compared to the same characteristics reported for other helicopter gunships in the US Army inventory. The results of the quantitative comparison are shown in Figures I and II attached. This comparison reveals that the AH-56A is as good or better than these aircraft which were all reported as having satisfactory handling qualities for the armed helicopter mission.

(b) The evaluation of the static lateral directional stability characteristics of the AH-56A was conducted under day, visual flight conditions only. The conclusion of deficient mission capability is based on projected characteristics under conditions not evaluated during these tests. The front seat pilot location; the non-standard instruments; instrument panel layout and controls; the lack of mission equipment; the lack of a trained copilot/gunner in the other seat and the lack of any simulated mission flight profiles in the test program are all significant factors in considering the validity of this conclusion. Based on the above, the static lateral directional stability characteristics are considered to be incorrectly classified and this deficiency should be deleted.

(c) It is agreed that improvement in the lateral directional stability characteristics of all helicopter gunships will enhance the mission capability under night and/or limited visibility conditions. Furthermore, a need exists for a design and evaluation specification for helicopters that encompasses all handling qualities under these operating conditions. Present planning calls for actual testing of the handling characteristics at airspeeds less than 100 KCAS under simulated instrument meteorological conditions. This testing will be accomplished using both the AH-56A man-in-the-loop simulator and the AMCS configured AH-56A aircraft to evaluate mission capability and requirements.

b. BLADE MOMENT STALL

(1) DEFICIENCY: Uncommanded aircraft motion and loss of control during maneuvering flight.

(2) ANALYSIS:

(a) This is an effect or consequence of entering blade moment stall. The combinations of conditions (gross weight, density altitude airspeed, collective blade angle, load factor, and the size and rate of cyclic control inputs) that define the boundary where blade moment stall will occur do not allow full utilization of the high airspeed, high-g capability of the AH-56A.

(b) Three (3) additional effects of the blade moment stall deficiency were included in this report as shortcomings. These are: "lack of satisfactory warning of uncommanded pitch-up," "inability to perform operational maneuvers at high airspeeds," "poor stick-fixed maneuvering stability." These three shortcomings should be declassified since they are directly related to the stated deficiency.

(c) While we concur that this characteristic is a deficiency, the production control configuration advanced mechanical control system (AMCS) will prevent the unacceptable aircraft reaction to blade moment stall by eliminating the rotor blade feathering feedback to the control gyro and will, therefore, correct this deficiency and related shortcomings. Two of the development aircraft are presently being modified to incorporate this control system and will be flight evaluated beginning September 1972.

c. ROTOR SPEED DECAY RATE

(1) DEFICIENCY: Rapid rate of rotor speed decay following simulated engine failures which allows the rotor to drop below the present transient limit.

(2) ANALYSIS:

(a) This is an invalid conclusion as reported. The "present transient minimum rotor speed" referenced in this report is not based on any known rotor or flying quality limit. The limit was suggested by the Contractor for flight test purposes because no lower rotor speeds had been evaluated during the envelope expansion tests preceeding this APE. Approaching or exceeding such limits does not substantiate classification as deficient unless a true aircraft or pilot limit is reached. No rotor or flying quality limits have been approached and reported during any of the autorotation entry tests conducted by the Army or Contractor which would seriously impair the safety of the crew aircraft.

(b) The minimum rotor speed experienced during this test was reported in Paragraph 116 as 91% well above the APE transient limit of 85%. The reported decay rates of 10-14% per second are not considered valid for extrapolating the minimum rotor speed after a two second delay because:

(1) simulated engine failure tests on other rotorcraft have shown that rotor speed decay is a non-linear function with respect to time; (2) decay rate after activation of the Delta Beta system (automatic pusher propellor blade angle reduction) will be reduced significantly.

(c) It is agreed that further testing is required to optimize the time and rate of activation and effect of the Delta Beta system and to evaluate the number, type and adequacy of pilot cues following sudden, complete engine failure for all flight conditions authorized. Thus, the characteristics reported do not constitute a deficiency which impairs safety or operational capability and should be declassified with the recommendation that further testing be conducted.

d. DIRECTIONAL CONTROL MARGIN

(1) DEFICIENCY: Inadequate directional control margin in sideward flight.

(2) ANALYSIS:

(a) Adequate or inadequate implies that there was a definition of the required performance in the applicable design, requirements or test specification. Since conditions of aircraft configuration, gross weight, density altitude were not defined, which would establish satisfactory mission capability, the classification of deficient is invalid. No actual or impending loss of aircraft control was reported because of this characteristic. At the most critical condition in sideward flight, as reported herein, sufficient directional control power is available to meet the requirements of paragraph 3.3.6, MIL-H-8501A. Therefore, this deficiency is invalid as developed in the report and should be declassified.

(b) The sideward flight performance has been improved to meet proposed new tactical profiles for anti-armor missions. A tail rotor control change was made subsequent to this test which provides 2° more tail rotor blade angle at the full left directional control position. The results of this change will be evaluated and reported by the Army during subsequent tests.

e. VIBRATION CHARACTERISTICS

(1) DEFICIENCY: Excessive 4/rev and 8/rev vibration levels.

AMCPM-AAH-TM-T

SUBJECT: Indorsement to Army Preliminary Evaluation I and Research and Development Acceptance Test I, AH-56A Cheyenne Compound Helicopter Final Report, March 1972

(2) ANALYSIS: The levels reported in the test aircraft, S/N 66-8834 are acknowledged as being high. The unique configuration of this aircraft (pilot's seat in the forward position containing a downward ejection seat and ballast to compensate for lack of nose and belly turrets), must be considered when evaluating the vibration characteristics of AH-56A aircraft. The vibration levels are reported as significantly lower in AH-56A S/N 66-8831, a more representative operational configuration. No impairment of the mission capability has been reported because of vibration levels during weapons testing accomplished in S/N 66-8831 and S/N 66-8832 by the Army, (Army Preliminary Evaluation II, III and Research and Development Acceptance Test I). Therefore, the classification of deficiency due to excessive vibration is considered invalid. No action is considered warranted since operational configurations have not been evaluated as deficient and none of the planned testing on aircraft S/N 66-8834 is impaired by this level of vibration.

4. In summary, four (4) of the five (5) deficiencies reported are incorrectly classified. The one deficiency remaining (Blade Moment Stall) must be corrected for operational acceptability. Continued development including incorporation of the production main rotor control system (Advanced Mechanical Control System) will correct this remaining deficiency. Testing scheduled during CY 73 will confirm this and the other improvements to the handling qualities and performance of the AH-56A.

2 Incl
as

HENRY H. BOLZ, JR.
Brigadier General, USA
Advanced Attack Helicopter
Project Manager

FIGURE [illegible]

COMPARISON OF ARMED HELICOPTER

STATIC LATERAL-DIRECTIONAL STABILITY

LEGEND	AIRCRAFT	AIRSPEED KTAS	GR. WT. LB.	DESN. ALT. FT.	NR RPM	C.G.
	AH-56A	59	18070	4700	248	AFT
	AH-1G	60	8665	4550	324	AFT
	UH-1C (XM-3/XM-5)	56	7840	5000	324	FWD
	UH-1C (XM-3/XM-5)	107	7890	5000	324	FWD

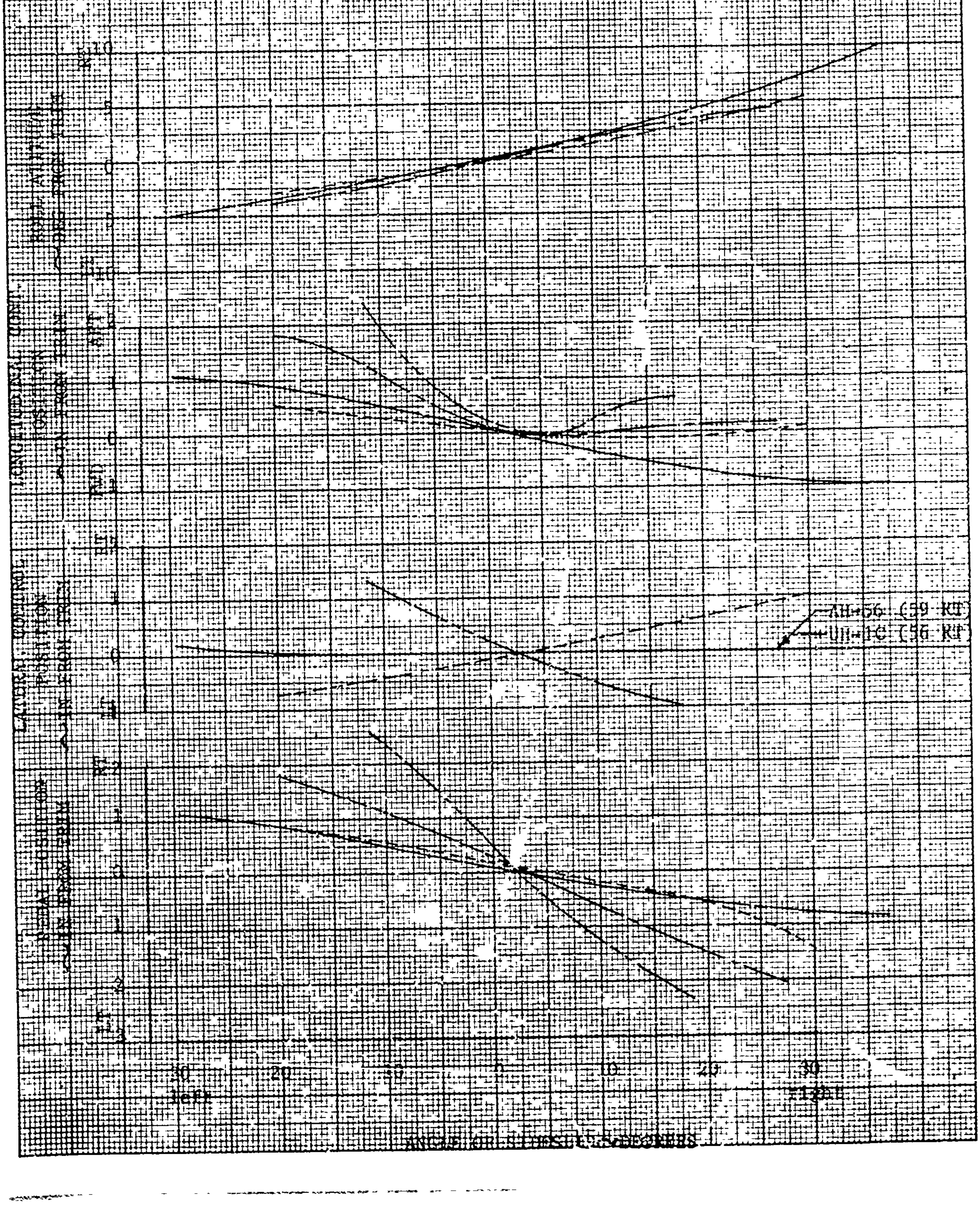

TABLE OF CONTENTS

INTRODUCTION

BACKGROUND

1. The AH-56A Cheyenne advanced aerial fire support system (AAFSS) is a major weapons system which is being developed by Lockheed Aircraft Corporation (LAC) under contract to the US Army Aviation Systems Command (AVSCOM). Design and development of the aircraft are being performed by LAC at its Lockheed-California Company (LCC) facility in Van Nuys, California. Contractor flight testing is being conducted at Yuma Proving Ground, Arizona.

2. The restructured AH-56A development program (ref 1, app A) identified two Army test programs which were to be conducted by the US Army Aviation Systems Test Activity (USAASTA) on the Improved Control System (ICS) configuration of the AH-56A. These programs were the Army Preliminary Evaluation I (APE I) and a portion of the Research and Development Acceptance Test I (RDAT I).

3. The APE I was divided into three phases (APE I.1, APE I.2, and APE I.3) to afford the Army the opportunity to assess the airworthiness of the AH-56A at various stages during the contractor development program. The APE I.1 was an evaluation of the handling qualities of the aircraft without external stores. The APE I.2 was an abbreviated evaluation of the external stores configuration. The APE I.3 was, primarily, a reevaluation of the handling qualities of the AH-56A with external stores following extensive main rotor and control system changes. Testing was also accomplished without external stores. This testing was accomplished by USAASTA personnel with data reduction assistance performed by the contractor.

4. The USAASTA portion of the RDAT I was performed to qualitatively evaluate the aircraft handling qualities during weapons firing. This test was accomplished by USAASTA personnel with copilot/gunner assistance from the US Army Aviation Test Board (USAAVNTBD).

5. Guidance for USAASTA participation in the AH-56A development program was provided by AVSCOM in references 2 through 7, appendix A. An approved test plan for APE I was first published in July 1970 (ref 8) and revised in February 1971 to include APE I.1 and APE I.2 (ref 9). The approved test plan for APE I.3 was published in July 1971 (ref 10). The approved RDAT I test plan was published in August 1971 (ref 11). Briefings and reports were submitted to AVSCOM following each test (refs 12 through 17). This report is a consolidated final report on the APE I.1, APE I.2, APE I.3, and RDAT I tests as directed by AVSCOM (ref 18).

TEST OBJECTIVES

6. The objectives of the APE I were as follows:

a. To provide an early assessment of the handling qualities of the reconfigured AH-56A development vehicle.

b. To assist in determining the flight envelopes to be used for future Army tests.

c. To obtain limited performance data for the modified AH-56A development vehicle.

d. To detect deficiencies which require correction.

e. To identify shortcomings for which correction is desired.

7. The objective of RDAT I was to determine the effects of weapons firing on the aircraft handling qualities.

8. Throughout all tests, emphasis was placed on airworthiness problem areas previously identified during the contractor development program and during Army tests. These areas of emphasis included:

a. Rotor dynamic stability characteristics.

b. Cross-coupling.

c. Lateral control migration.

d. Crew station vibration environment.

e. Directional control power.

f. Blade moment stall.

g. Pilot-coupled roll oscillations.

DESCRIPTION

9. The AH-56A is a compound helicopter designed to perform the advanced aerial fire support mission. In addition to a single four-bladed "rigid" main rotor and a teetering four-bladed antitorque tail rotor, a three-bladed pusher propeller is located at the aft end of the fuselage, and a low wing is located on the fuselage midsection. During high-speed forward flight, the main rotor is partially unloaded with the lift provided by the wing and thrust supplied by the pusher propeller. The cockpit has tandem seats. Normally, the forward seat is for the copilot/gunner,

and the aft seat is for the pilot. Provisions for a wide variety of armament systems are available in two turrets and on six external stores stations. The conventional wheel-type landing gear is retractable. Power is provided by a single General Electric T64-GE-16 turboshaft engine which has a maximum rating of 3925 shaft horsepower (shp) at sea-level (SL), standard-day conditions.

10. Two test aircraft were employed: the APE I was conducted using AH-56A serial number (S/N) 66-8834 (LCC S/N 1009), and the RDAT I was conducted using AH-56A S/N 66-8831 (LCC S/N 1006). Both aircraft were configured with the ICS and a reverse rotation tail rotor (clockwise when viewed from the left side of the aircraft).

11. The APE I aircraft is being used by the contractor for aerial vehicle flight development and is configured with a downward ejection seat installed in the forward cockpit, the pilot station for USAASTA tests, in lieu of the swivelling gunner station. The aircraft did not have the standard XM52 30mm belly turret or the external swivelling sight head installed. A simulated XM51 40mm nose turret was installed. Significant configuration changes during the preparation for and during the conduct of this test were as follows:

a. For APE I.1, the primary changes to the aircraft from the earlier development models included: increased forward blade sweep, addition of main rotor blade tip weights, reduced main rotor inplane natural frequency, increased control gyro arm stiffness, reduction of main rotor delta-three coupling, increased collective control stiffness, increased collective servo capacity, installation of a roll desensitizer, and addition of a collective control brake. The basic weight of the test aircraft with instrumentation installed was 14,036 pounds with the center of gravity (cg) at fuselage station (FS) 313.4.

b. For APE I.2, the aircraft was configured as during APE I.1, except that empty tube-launched, optically tracked, wire-guided (TOW) missile pods were installed on each inboard stores station, and loaded XM159 rocket pods were installed on each outboard stores station.

c. Major configuration changes made between APE I.2 and APE I.3 included: increased main rotor droop; installation of a roll compensator, pitch desensitizer, and pitch/roll decoupler; increased design effectiveness of the bobweight; alteration of right wing and horizontal stabilizer incidence; removal of detuning weights from the horizontal stabilizer; and removal of the roll desensitizer. The APE I.3 was conducted with and without loaded TOW missile pods installed on each inboard stores station and loaded XM159 rocket pods on each outboard stores station. The basic weight of the test aircraft with instrumentation installed was 14,656 pounds with the cg at FS 312.3.

d. During the conduct of APE I.3, the aircraft was returned to the contractor for a series of configuration modifications. As a result, APE I.3 was divided into APE I.3A and APE I.3B to denote premodification and postmodification testing, respectively. These modifications included: addition of roll compensator notch filters at 16 and 32 hertz (Hz), activation of the lift/roll decoupler, reduction of control system free play (not a design change), increased lateral control travel and reduced lateral control effectiveness, reduced lateral control breakout force, increased swashplate feedback, addition of detuning weights to the right wing, and detuning of the collective control lever in the aft cockpit.

12. The RDAT I vehicle is being used by the contractor for development and demonstration of the fire control and weapons systems. During the USAASTA testing, the rotor and control system were configured as described in paragraph 11c. The weapons configuration for the RDAT I included one XM51 (40mm) weapon in the chin turret, one XM52 (30mm) cannon in the belly turret, one XM159 rocket pod on each outboard wing store station, and one simulated TOW missile pod on each inboard wing store station. The basic weight of the aircraft with instrumentation installed was 16,017 pounds with the cg at FS 303.7.

13. A more detailed description of the AH-56A is given in appendix B. Configuration differences peculiar to each USAASTA test are specified.

SCOPE OF TEST

14. The USAASTA testing was confined to contractually defined productive test hour and elapsed calendar time limits. The developmental nature of the AH-56A aerial vehicle program required emphasis on certain of the preplanned test objectives. These factors combined to limit accomplishment of planned testing. As a result, no performance and limited autorotational entry tests were accomplished. The resulting scope of test is shown in table 1.

Table 1. Scope of Test.

Test	Calendar Days Charged to USAASTA Testing[1]	Number of Test Flights	Number of Productive Test Hours
APE I.1	18	18	15.2
APE I.2	7	3	1.0
APE I.3A	32	16	7.5
APE I.3B	23	23	15.5
RDAT I	10	5	4.3
Total:	90	65	43.5

[1]Excludes pilot training and refamiliarization time and periods during which aircraft was unavailable to USAASTA. Includes weekend time and other days during which no USAASTA flying occurred due to administrative, weather, and maintenance delays.

Table 2. Test Conditions.[1]

Test	Type of Test	Nominal Test Airspeeds[2] (KCAS)	Nominal Density Altitude (ft)	Nominal Gross Weight and External Configuration	Nominal Center of Gravity
APE I.1	Hover, takeoff, and landing	N/A	4000	18,300 pounds Clean	FS 300 (mid)
	Control positions in trimmed forward flight	50 to 190			
	Static longitudinal stability	60, 100, 150, and 180			
	Static lateral-directional stability	60, 100, 150, and 180			
	Dynamic stability	0, 60, 100, 150, 165, and 180			
	Controllability	0, 60, 100, 150, 165, and 180			
	Maneuvering stability	80, 155, and 180			
APE I.2	Static longitudinal stability	150	4000	18,300 pounds External stores	FS 300 (mid)
	Static lateral-directional stability	60 and 100			
	Maneuvering stability	80, 150, and 180			
APE I.3	Sideward and rearward flight	N/A	750	18,300 pounds Clean and 20,000 pounds External stores	FS 300 (aft)[3] and FS 292 (fwd)[4]
	Static longitudinal stability	60, 100, 150, 180, and 190	5000		
	Static lateral-directional stability	60, 100, 150, 170, 180, and 195			
	Dynamic stability	0, 60, 100, and 150			
	Controllability	0, 60, 100, and 150			
	Maneuvering stability	80, 120, 150, 170, and 180			
	Autorotational entry	150, 160, and 180			
RDAT I	XM51 (40mm nose turret) firing	0, 60, and 150	4000	20,000 pounds External stores	FS 300 (aft)[3]
	XM52 (30mm belly turret) firing	60, 150, and 180			
	XM159 (2.75 in. rocket) firing	120, 140, and 180			

[1]Rotor speed: 246 rpm (100 percent).
[2]Not all variables tested at all weights, configurations, and speeds.
[3]Envelope redefined following APE I.2 such that FS 300 is aft cg.
[4]Limited testing accomplished at forward cg.

15. All testing was conducted at Yuma Proving Ground, Arizona. Limitations established by the AH-56A safety-of-flight releases issued by AVSCOM were observed (app C). Basic test conditions are listed in table 2.

METHODS OF TEST

16. The methods of test used are established engineering flight test techniques and are briefly described in the Results and Discussion section of this report. Test results are compared to the applicable requirements of the development description CP0001A (ref 1, app A) and military specifications MIL-H-8501A (ref 19) and MIL-F-8785(ASG) (ref 20). A Handling Qualities Rating Scale (HQRS) (app D) was used during evaluation of mission tasks as defined in reference 21, appendix A.

17. For the APE I testing, flight test data were obtained from instrumentation located on the photopanel and two oscillographs in aircraft S/N 66-8834. The RDAT I test aircraft, S/N 66-8831, was instrumented primarily for weapons firing and lacked handling qualities instrumentation. A detailed listing of the APE I test instrumentation is included in appendix E.

CHRONOLOGY

18. The chronology of the AH-56A test program is as follows:

Ground school training completed	18	December	1970
APE I.1 and APE I.2 test directive received	9	January	1971
APE I.3 and RDAT I test directive received	27	January	1971
APE I.1 flight training completed	29	January	1971
APE I.1 testing initiated	30	January	1971
APE I.1 testing delayed for congressional observation of test	31	January	1971
APE I.1 testing resumed	10	February	1971
APE I.1 testing completed	5	March	1971
APE I.3 revised test directive received	5	March	1971
APE I.1 debriefing for AVSCOM, the Advanced Aerial Weapons Systems Project Manager (AAWS-PM), and LCC	12 - 13	March	1971
APE I.1 test report submitted to AVSCOM	2	April	1971
APE I.2 testing initiated	23	April	1971
APE I.2 testing completed	29	April	1971
APE I.2 debriefing for AVSCOM, AAWS-PM, and LCC	30	April	1971
RDAT I revised test directive received	20	May	1971

APE I.2 test report submitted to AVSCOM	27	May	1971
APE I.3 flight training completed	9	September	1971
APE I.3A testing initiated	10	September	1971
Roll compensator test directive received	28	September	1971
RDAT I flight training completed	28	September	1971
RDAT I testing initiated	29	September	1971
Roll compensator evaluation initiated	5	October	1971
RDAT I testing completed	8	October	1971
Roll compensator evaluation terminated	11	October	1971
Roll compensator debriefing for AVSCOM, AAWS-PM, and LCC	12	October	1971
APE I.3A testing terminated	19	October	1971
RDAT I test report submitted to AVSCOM	17	November	1971
APE I.3B testing initiated	2	December	1971
APE I.3B testing completed	23	December	1971
APE I.3 debriefing for AVSCOM and AAWS-PM	11	February	1972
APE I.3 debriefing for LCC	22	February	1972

RESULTS AND DISCUSSION

GENERAL

19. The flying qualities evaluation of the AH-56A conducted during APE I and the USAASTA portion of RDAT I revealed five deficiencies and 54 shortcomings which exist within the presently approved test flight envelope. The handling qualities were not significantly affected by firing of any of the weapons systems tested. The deficiencies, correction of which should be a prerequisite for an airworthiness release for operational Army aviators, are (1) excessive pilot workload due to unacceptable static lateral-directional stability characteristics below 100 knots seriously impairs the capability to operate at minimum altitudes unaffected by conditions of darkness or adverse weather, (2) uncommanded aircraft motion and loss of control during maneuvering flight, (3) rapid rate of rotor speed decay following simulated engine failures which allows the rotor speed to drop below the present transient limit, (4) inadequate directional control margins in sideward flight, and (5) excessive four-per-rotor-revolution (4/rev) and 8/rev vibrations. Deficiencies (2) and (4) above warrant a reduction of the flight envelope size for future Army tests until correction of these deficiencies is accomplished. Lateral stick migration with airspeed was not objectionable, the pusher propeller provided excellent deceleration capability, and the ability to control airspeed independent of dive angle is an excellent feature.

FLYING QUALITIES

General

20. The results of flying qualities testing are presented in this section. Quantitative test data were gathered during APE I.1, APE I.2, APE I.3A, and APE I.3B. The configuration of the test aircraft varied significantly for each test phase (para 11). To simplify identification of test data, the data are presented separately as follows: appendix F contains APE I.1 data, appendix G contains APE I.2 data, appendix H contains APE I.3A data, and appendix I contains APE I.3B data. Qualitative data were obtained during both APE I and the USAASTA portion of RDAT I. Unless otherwise identified, the test results presented in this report are representative of the APE I.3B aircraft configuration.

Control Positions in Trimmed Forward Flight

21. Control position requirements for stabilized trimmed forward flight were determined during APE I.1 for the clean configuration at an 18,400-pound gross weight. Data were obtained by stabilizing the aircraft in trimmed zero sideslip flight at 10-knot speed increments from 50 to 162 knots calibrated airspeed (KCAS) in level flight and with smaller increments in diving flight at 172 KCAS and above. From 80 KCAS to the maximum airspeed for level flight at maximum continuous

power (V_H), main rotor collective pitch was maintained at the maximum value permitted by appendix C, and propeller pitch (Beta) was added as necessary to maintain a constant altitude. At airspeeds faster than V_H, Beta was held constant at 35.5 degrees, and the aircraft was dived to obtain speeds to 190 KCAS. Below 80 KCAS, where collective angle was not limited, Beta was held approximately constant at 17 degrees, and altitude was maintained with collective. Control positions in trimmed forward flight are shown in figure 1, appendix F.

22. The longitudinal control required for trim was essentially constant at low airspeeds (50 to 75 KCAS) where the collective was used for altitude control. Between 80 and 120 KCAS, essentially linear increases in forward control position were required for increasing airspeed. The requirement for additional forward longitudinal control diminished at higher speeds, and in diving flight above V_H, only very small longitudinal trim changes were associated with changes of trim speed. Two-degree variations in collective pitch from 5 degrees produced approximately 0.3-inch longitudinal trim changes. Increased collective pitch required a forward longitudinal trim change.

23. Left lateral control displacement was required to maintain trimmed flight at all forward airspeeds tested. The lateral control position varied essentially linearly from 1 inch left of center at 70 KCAS to 0.4 inch left of center at 170 KCAS. Neither the lateral control displacement nor its variation with airspeed (lateral stick migration with increasing forward airspeed) was objectionable at airspeeds above transition. Transitional characteristics are discussed in paragraph 97.

24. Directional control position remained essentially constant between 70 and 190 KCAS. The rigging was such that the right pedal was approximately 2 inches forward of the left pedal at these speeds. This pedal position was uncomfortable, contributed to difficulty in maintaining zero sideslip, and is a shortcoming.

25. Separate tests were not conducted during APE 1.2 and APE 1.3 to explicitly determine trim characteristics at higher gross weights and with the external stores configuration. However, examination of the trim points obtained during static longitudinal stability and static lateral-directional stability tests indicates that gross weight and external stores have a negligible influence on control positions in trimmed forward flight. The effects of landing gear extension and retraction on forward flight trim requirements were negligible.

Trimmability

26. The AH-56A has trim motors in the lateral, longitudinal, and directional control systems. An electric friction brake is incorporated in the collective pitch control system to augment the adjustable mechanical friction. This brake is disengaged when the landing gear is extended. Lateral and longitudinal trim changes are made with a "coolie hat" switch on the cyclic stick. Directional trim changes are made using a switch located on the collective pitch lever.

27. Longitudinal trim was difficult at airspeeds above 150 KCAS even though the functioning of the longitudinal trim system was satisfactory. At these

conditions, the pilot established the desired pitch attitude, the aircraft appeared stable, and the pilot trimmed the control forces to zero. The control position generally was not that required for the desired pitch attitude because of the long pitch time constant (para 64). This resulted in the aircraft deviating from the desired attitude which required additional control inputs and retrimming. Longitudinal trim required excessive pilot attention at high airspeeds and is a shortcoming.

28. Lateral trim was difficult at all airspeeds tested because of an apparent tendency for the trim system to overshoot. Trim was finally attained only after several corrections were made back and forth. Lateral trim required excessive pilot attention and is a shortcoming.

29. Directional trim was satisfactory in forward flight, but was not adequate to cope with the large, rapid changes that take place while accelerating or decelerating through the transition regime (paras 97 and 98). The rate of operation of the directional trim motor was too slow to correct for the trim requirements during transition, resulting in a need to hold large directional control forces to maintain a constant heading. Compounding the problem was the location of the directional trim switch on the collective lever. Undue mental concentration was required to relate left thumb movement to foot pressures. At the same time, the left hand was operating collective and propeller pitch to control aircraft height and acceleration. These characteristics of the directional trim system increase the workload in a high-workload flight regime (para 112). The excessive pilot attention required to trim directionally during hover transitions is a shortcoming.

30. The friction devices incorporated in the collective control system were adequate to maintain the desired position with the landing gear retracted and the electric brake activated. However, with the gear down and the electric brake disengaged, the collective lever would creep, changing the collective blade angle, unless the mechanical friction was tightened fully. The amount of collective friction applied cannot be determined by either feeling or looking at the adjustment control. These unsatisfactory collective friction characteristics are a shortcoming.

31. Control of propeller blade angle required continual pilot attention when operating at high blade angles (above 30 degrees). The blade angle crept to a lower setting, although the friction was fully tightened. Reduction of the blade angle degraded the ability to hold airspeed and altitude constant because of the thrust loss. The inability of the propeller control system friction to maintain the desired settings is a shortcoming.

Static Longitudinal Stability

32. Collective-fixed static longitudinal stability was evaluated during APE 1.1, APE 1.2, and APE 1.3. Trim speeds from 60 to 189 KCAS were examined in the clean configuration. The effects of external stores configuration, gross weight, collective pitch, and cg were examined at selected trim airspeeds. During this test, collective and propeller blade angles were held constant at the settings required

to maintain the trim condition. Airspeed was stabilized at increments above and below the trim point by varying longitudinal control position. Zero sideslip was maintained by varying lateral and directional controls. Test data are presented in figures 2 through 5, appendix F; figure 1, appendix G; figures 1 through 7, appendix H; and figures 1 through 4, appendix I.

33. During APE I.1, the longitudinal control position stability was positive at low airspeed and decreased with increasing airspeed. At approximately 10 knots above or below the 150-knot trim point, the control position stability became neutral to negative. At 177 KCAS, the stability became neutral to negative approximately 5 knots above trim airspeed.

34. During APE I.1, the longitudinal control force stability about trim was positive at 100 and 150 KCAS and was reduced at both higher and lower airspeeds. Above 100 KCAS, airspeed variations of 10 knots or more from trim resulted in reduced longitudinal control force stability.

35. Configuration changes made between APE I.2 and APE I.3 resulted in generally degraded static longitudinal stability. During APE I.3, at airspeeds above 150 KCAS, the stability increased with increasing speed (opposite APE I.1 trends, para 33); however, weak static longitudinal control force and control position stability were still apparent at these airspeeds. At trim speeds of 100 and 150 KCAS, the longitudinal control force stability for APE I.3 was substantially reduced from APE I.1 levels.

36. Reduced collective pitch caused a pronounced degradation in static longitudinal stability. Data obtained with 3.7 degrees of collective at 149 and 189 KCAS trim speeds demonstrated neutral-to-negative control position and control force stability. The effects of external stores configuration, gross weight, and cg could not be established from the limited data but appeared to be small. During the static longitudinal stability tests, minor trim changes with airspeed were noticed in the lateral and directional axes.

37. The static longitudinal stability characterisitcs of the AH-56A do not meet the requirements of paragraph 3.2.10 of MIL-H-8501A, in that the gradients of control position and control force with airspeed are not stable for all flight conditions. However, the apparent speed stability of the AH-56A is considerably better than evidenced by those gradients, because of the airspeed stabilization provided by the propeller. The propeller thrust (with blade angle fixed) increases with decreased speed and decreases with increased speed. Hence, it was very easy to maintain a trim airspeed.

38. The weak static longitudinal stability degraded pitch attitude control. Whenever the pitch attitude was disturbed, by a gust or a control movement, there was very little tendency for the aircraft attitude to return to trim. Pilot attention was then required to return the aircraft to the desired attitude in order to maintain constant airspeed and altitude (HQRS 4). The weak static longitudinal stability at 150 KCAS and above is a shortcoming.

Static Lateral-Directional Stability

39. Static lateral-directional stability characteristics were evaluated during APE I.1, APE I.2, and APE I.3. Trim speeds from 59 to 180 KCAS for the clean configuration and from 61 to 172 KCAS for the external stores configuration were evaluated. The static lateral-directional tests were conducted by trimming the aircraft at a selected airspeed and zero sideslip. Main rotor collective and propeller blade angles, airspeed, force trim settings, and aircraft ground track were held fixed. Sideslips were increased incrementally, both left and right, up to the flight envelope limits. Test data are presented in figures 6 through 9, appendix F; figures 2 and 3, appendix G; figures 8 through 12, appendix H; and figures 5 through 7, appendix I.

40. The static directional stability, as evidenced by the variation of directional control position with sideslip, was positive about zero sideslip at all speeds tested but was very weak at low speed. External stores had no significant influence on directional stability. The directional control gradient about trim increased essentially linearly with airspeed. At trim speeds of 150 KCAS and below, the directional stability became degraded as sideslip angles were increased, and at 150 KCAS, did not comply with paragraph 3.3.9 of MIL-H-8501A which requires pedal displacement with sideslip to be approximately linear between ±15 degrees of sideslip. The test flight envelope (app C) did not permit investigation of this requirement at higher airspeeds. Weak-to-neutral directional stability was encountered at 61 KCAS and approximately 30 degrees of sideslip (fig. 2, app G) and also at 99 KCAS and approximately 20 degrees of right sideslip (fig. 7, app F). The data obtained indicate that there was no siginificant change in directional stability between APE I.1 and APE I.3. Paragraph 3.3.9 of MIL-H-8501A requires positive directional stability for all sideslips and airspeeds from 45 degrees at 50 knots varying linearly to 15 degrees at maximum allowable airspeed (V_{max}). Lack of adequate directional stability made it very difficult to establish and maintain a desired heading and sideslip and is a shortcoming.

41. The dihedral effect, as evidenced by the variation of lateral control position with sideslip during APE I.1, was essentially zero near 60 KCAS and increased to slightly positive values at higher airspeeds. External stores had no significant effect on dihedral effect. Prior to APE I.3B, the lateral control system was modified to provide reduced control power (control moment per inch of control displacement) and increased control travel. As a result, the apparent dihedral effect was nearly doubled at high speeds and was satisfactory about trim. Near 60 KCAS, the dihedral effect was essentially unchanged and was neutral to negative throughout the sideslip angles tested. The dihedral effect did not meet the requirements of paragraph 3.3.9 of MIL-H-8501A.

42. In general, the side-force characteristics, as indicated by the variation in bank angle with sideslip, varied from slightly positive near 60 KCAS to strongly positive at 180 KCAS. Howerver, data obtained during APE I.2 (external stores) indicated that the side-force gradient became weak to neutral prior to reaching the sideslip envelope limit. This characteristic is highly undesirable. Good side-force

characteristics are desirable as this is a primary cue of sideslip, especially during instrument or low-visibility conditions when reference to attitude and ground track are difficult. Side force is also the parameter indicated by the ball of the needle-ball (turn-and-bank) indicator. Previous testing (ref 22, app A) indicates that a side force corresponding to 10 percent of the normal acceleration is required for positive recognition of sideslip. This side force would be 0.1g for 1g flight which corresponds to a bank angle of 5.7 degrees. During APE I.3 testing with no external stores, the sideslip angle corresponding to 5.7 degrees of bank angle was approximately 30 degrees at 59 KCAS and decreased to approximately 8 degrees, left, and 3 degrees, right, at 180 KCAS. At airspeeds of 100 KCAS and above, the sideslip envelope to the left would be exceeded prior to positive recognition of sideslip. At airspeeds of 150 KCAS and above, positive recognition of sideslip to the right would probably occur approximately 1 degree before reaching the envelope limit. The weak side force at low airspeeds, the side-force characteristics with respect to the limited sideslip envelope at airspeeds of 100 KCAS and above, and the degraded side-force characteristics at higher sideslip angles with external stores combine to make the side-force characteristics of the AH-56A a shortcoming.

43. Inherent sideslip (sideslip angle for ball-centered flight) changed significantly between APE I.2 and APE I.3. During APE I.1 and APE I.2, the bank angle at zero sideslip was approximately zero (wings level) throughout the airspeed range tested. During APE I.3, however, the bank angle for zero sideslip varied from zero degrees at 60 KCAS to approximately 2 degrees (right wing low) at 180 KCAS, which was uncomfortable. This change in bank angle is probably due to the right wing incidence change made prior to APE I.3. The primary cue of sideslip and the parameter sensed by the slip ball is side force. Lacking a direct sideslip indication, the pilot will tend to fly at zero side force (wings level in straight flight). At airspeeds of 100 KCAS and above, zero side force corresponds to approximately 3 degrees of left sideslip. During this test, the sideslip envelope was exceeded at high airspeeds even though a sensitive sideslip indicator was installed. Because of the inherent sideslip and the side-force characteristics, the small sideslip envelope at high speeds (4 degrees at 200 KCAS) will be exceeded. This inherent sideslip will degrade rocket firing accuracy, may affect performance, and is a shortcoming.

44. During the lateral-directional tests, a very strong longitudinal trim shift with sideslip (pitch due to sideslip) was encountered. Forward longitudinal control displacement was required to counteract the nose-up trim shift experienced in right sideslips, and aft longitudinal control was required to counteract the nose-down trim shift in left sideslips. This characteristic was also encountered during maneuvering and controllability testing and is further discussed in paragraphs 57 and 77. The magnitude of longitudinal and directional control displacements from trim were approximately equal. The pitch with sideslip required excessive pilot compensation to achieve a desired combination of sideslip and airspeed. Sideslip frequently resulted from directional changes and from maneuvering. The pitch coupling with sideslip which resulted required considerable pilot attention. This was particularly objectionable when rapid target changes were made while simulating an attack using the flexible weapons in the stowed mode or an attack using rockets.

The resulting deviations of the vertical sight line required several seconds to align the sight on the target. Also, this pitch-with-sideslip coupling makes airspeed control difficult when making small pedal-only heading changes as is normally done during instrument flight. External stores had no significant effect on the longitudinal trim shift with sideslip. The gradient of longitudinal control displacement versus sideslip increased with increasing airspeed, was objectionable at all forward speeds tested, and is a shortcoming.

45. The forward longitudinal control force during right sideslips exceeded the 3-pound limit of paragraph 3.3.20 of MIL-F-8785(ASG). That specification paragraph also indicates preference for increasing pull force accompanying increasing sideslip and that the magnitude and direction of the trim change, if present, be similar for right and left sideslips. The AH-56A fails to meet this specification requirement. The excessive longitudinal control force, particularly push forces, required during sideslips was objectionable to the pilot and is a shortcoming.

46. The ability to recognize and maintain a desired sideslip is required to remain within the present stores jettison envelope which restricts stores jettison to zero sideslip and airspeeds below 100 KIAS. Additionally, sideslip degrades folding-fin aircraft rocket (FFAR) firing accuracy. During takeoffs and landings and target attack missions, the ability to control ground track and aircraft heading is required. During all tests, difficulty was experienced in obtaining and maintaining the desired sideslip even though a sideslip indicator was installed in the aircraft.

47. Operation of the AH-56A under night or low-visibility conditions while performing nap-of-the-earth or low-level flight, must be done at reduced airspeeds in order to provide sufficient time for obstacle identification and avoidance. In the airspeed regime below 100 knots, the shortcomings of inadequate directional stability (para 40), neutral-to-negative dihedral effect (para 41), and weak side-force characteristics (para 42) combine to require excessive pilot attention for adequate aircraft control. This requirement for excessive pilot attention due to the unacceptable static lateral-directional stability characteristics below 100 KIAS seriously impairs the capability to operate at minimum altitudes unaffected by conditions of darkness or adverse weather and is a deficiency.

<u>Maneuvering Stability</u>

General:

48. Two techniques were used to evaluate the maneuvering stability characteristics of the AH-56A: left and right windup turns and symmetrical pull-ups and pushovers. For both techniques, the aircraft was first stabilized at a trim airspeed in level flight, and the collective, propeller blade angle, and force trim settings were maintained throughout the maneuver. During windup turns, the aircraft was stabilized at increasing increments of normal acceleration in a constant airspeed turn. Because of pitch due to sideslip (para 44), sideslip angles in the windup turn were kept at a minimum with the aim condition being zero sideslip. Longitudinal control force and position data were corrected to zero sideslip using

the results of the static lateral-directional stability tests. The symmetrical pull-ups and pushovers were performed by diving and climbing the aircraft, establishing either decreasing or increasing increments of normal acceleration as the aircraft passed through the level flight attitude at the trim airspeed and altitude. During the pull-ups and pushovers, the aircraft was not stabilized, and therefore the results of these tests are qualitative. The results of the maneuvering stability tests are presented in figures 10 through 14, appendix F; figure 4, appendix G; figures 13 through 19, appendix H; and figures 8 through 17, appendix I. Table 3 presents the maneuvering stability test conditions.

Stick-Fixed Maneuvering Stability:

49. During APE I.1, the stick-fixed maneuvering stability (longitudinal stick position/load factor) of the test aircraft was generally poor. At 83 KCAS, the stick position gradient (change in longitudinal stick position/change in load factor) was approximately 3.7 inches per g (in./g) and was linear, a desirable characteristic. However, at 155 KCAS, the stick position gradient decreased with increasing load factor and became neutral at approximately 1.75. At 178 KCAS, the stick position gradient was very shallow about trim and became neutral at approximately 1.5g. These characteristics did not meet the requirements of paragraph 3.3.4 of MIL-F-8785(ASG) which requires that the slope of the curve of the longitudinal stick position versus normal acceleration at constant speed be stable (increasing aft longitudinal cyclic stick required for increasing load factor) throughout the attainable load factors in all configurations and in all conditions of flight. There was no significant difference between longitudinal stick position gradients obtained in right and left windup turns at all airspeeds tested.

50. During APE I.2, the stick-fixed maneuvering stability was evaluated at 150 KCAS in the external stores configuration (fig. 4, app G). The stick-fixed maneuvering stability appeared to be slightly degraded from APE I.1 tests. The gradient about trim decreased from 1.3 in./g in the clean configuration to 1.0 in./g in the external stores configuration. The load factor at which the gradient became neutral decreased from approximately 1.75 to approximately 1.7. These changes were not necessarily due to the addition of external stores, since the cg was 1 inch further aft than during the APE I.1 (which normally tends to decrease the maneuvering stability) and the airspeed was 5 knots slower.

51. The poor stick-fixed maneuvering stability was essentially unchanged from APE I.1 to APE I.3. The data were insufficient to determine the effects on the stick-fixed maneuvering stability of variations of external stores configuration, longitudinal cg position, collective blade angle, aircraft gross weight, or type of maneuver. The stick-fixed maneuvering stability fails to meet the requirements of paragraph 3.3.4 of MIL-F-8785(ASG) and is a shortcoming. Improvement of the longitudinal control characteristics to provide a constant or increasing control displacement per g with increasing airspeed and load factor is desired.

Table 3. Maneuvering Stability Test Conditions.

Test	External Configuration	Gross Weight (lb)	Center-of-Gravity Location (in.)	Density Altitude (ft)	Trim Airspeed (KCAS)	Maneuver Windup Turn	Collective Angle (deg)
APE I.1	Clean	18,070	299.7	3280	86	Left	7.2
		18,120	299.6	3380	80	Right	7.2
		18,200	299.4	3880	154	Left	5.0
		18,650	299.8	4200	156	Right	5.1
		18,180	299.6	3080	178	Left	5.1
		18,040	299.6	3340	178	Right	5.1
APE I.2	External stores	18,110	300.3	3770	150	Left	5.2
		17,930	300.8	3540	150	Right	5.2
APE I.3A	External stores	19,500	299.6	5370	81	Left	7.6
		18,800	300.0	3290	120	Left	5.4
		19,870	299.0	5180	151	Left	3.4
	Clean	17,990	299.0	6210	150	Left	5.8
	External stores	19,870	299.9	5680	152	Left	5.8
	Clean	17,790	299.0	5210	180	Left	5.3
		17,630	299.9	5130	181	Right	5.5
	External stores	19,350	297.3	5210	181	Left	3.5
APE I.3B	Clean	18,300	301.2	4840	80	Left	7.6
		18,420	300.1	4630	119	Left	5.8
		18,530	300.2	4570	151	Left	3.8
		18,230	300.1	5400	150	Left	5.8
		18,250	293.1	5500	149	Left	5.8
		17,870	300.0	5500	152	Right	3.8
	External stores	20,210	299.9	4100	151	Left	5.5
	Clean	18,410	300.2	3750	182	Left	3.8
		18,040	300.1	5890	182	Left	5.8
		18,040	293.0	4660	181	Left	5.5
	External stores	20,050	299.8	3800	182	Left	5.5
RDAT I	External stores	18,500	N/A	4000	145	Left	N/A
		18,500	N/A	N/A	160	Pull-up & pushovers	N/A
		18,500	N/A	N/A	185	Pull-up & pushovers	N/A

Stick-Free Maneuvering Stability:

52. During APE I.1, the stick-free maneuvering stability (longitudinal stick force/load factor) was adequate at low airspeeds and poor at the higher airspeeds. At 83 KCAS, the stick force gradient was approximately 25 pounds per g (lb/g) about the 1.0g trim point. This value decreased with increasing load factor to approximately 13 lb/g at the envelope limit of 1.5g. The 25-lb/g stick-force gradient is slightly in excess of the maximum permitted by CP0001A. The stick-free maneuvering stability decreased with increasing airspeed and load factor. At 155 KCAS, it became very weak to neutral at approximately 2.0g, which was the limit of the flight envelope. At 178 KCAS, the stick-free maneuvering stability became neutral at 1.6g and negative at higher load factors. The stick-free maneuvering stability gradients failed to meet the requirements of CP0001A and paragraph 3.3.9 of MIL-F-8785(ASG) because the gradients were excessively nonlinear and decreased with increasing load factor. During APE I.2, it was determined at 150 KCAS that the addition of external stores did not affect the stick-free maneuvering stability.

53. The extensive control system modifications following APE I.2 produced some change in the stick-free maneuvering stability. This was primarily due to the increased bobweight design effectiveness. The APE I.3 data showed that at 80 and 120 KCAS the gradient increased with increasing load factor up to the maximum load factors tested (1.5 and 1.6, respectively). As airspeed increased, the gradient about trim decreased. Above 150 KCAS, the force gradient decreased with increasing load factor. There was, however, an increase in the gradients above 150 KCAS when compared with APE I.1 data. Even with the desirable increased bobweight design effectiveness, the stick-free maneuvering stability is degraded by nonlinearities, which contributed to the deficient maneuvering characteristics described in paragraph 61.

Lift/Roll Coupling:

54. The APE I.1 testing at 155 KCAS revealed large left lateral stick displacement and force required with increased load factor (lift/roll coupling). At the 155-KCAS trim speed, the lateral stick displacement from trim was approximately 1.5 inches at 2.0g which represented approximately 27 percent of the total stick travel and corresponds to approximately 36 deg/sec of right roll rate. During rapid pull-up maneuvers, the lift/roll coupling appeared more severe because of the rapid requirement for left lateral stick movement while attempting to maintain wings level as the load factor increased. During pushovers, a pronounced and rapid left roll occurred as well as a very pronounced increase in 4/rev vibration. The APE I.2 testing showed no change in the lift/roll coupling with the addition of external stores.

55. The APE I.3B testing showed a significant reduction in lift/roll coupling by the addition of the lift/roll decoupler. The decoupler senses increasing load factor and airspeed and applies control inputs through a modulation (mod) piston in the roll servo. The roll moments produced by these inputs oppose the moment created

by the lift/roll coupling. During pull-ups and pushovers, lift/roll coupling was much less than previously encountered. There was, however, some residual roll in the aircraft which appeared to be caused by the phasing and gain of the decoupler. During pull-ups, a left roll resulted from an increase of load factor followed by a right roll as load factor was sustained. This characteristic resulted in an apparent roll oscillation when load factor was varying, complicating the maneuvering task. This characteristic was most objectionable at 150 KCAS since the coupling was the largest at that airspeed.

56. During APE I.3B, the lift/roll coupling was still objectionable because the large lateral displacements and low lateral forces were disharmonious in relation to the small longitudinal displacements and large longitudinal forces which occurred during high-speed maneuvering. Determination of the proper combination of control displacements and forces was an excessively difficult task which was aggravated by changes due to airspeed and load factor. Lift/roll coupling and the lack of consistent longitudinal and lateral control harmony are shortcomings.

Pitch Due to Sideslip:

57. Pitch-due-to-sideslip coupling (paras 44 and 77) complicated the pilot's task while trying to stabilize at desired load factors. The pitch changes introduced by sideslip angles in maneuvering flight caused perturbations in the normal load factor. Due to the lift/roll coupling, these small changes in load factor caused the aircraft to oscillate in roll. These roll excursions precluded precise control of the aircraft and were objectionable. On one occasion, pitch due to sideslip resulted in an increase of load factor sufficient to cause blade moment stall (fig. 20, app H).

Blade Moment Stall:

58. Uncommanded nose-up pitch rates occurred while attempting to stabilize at load factors within the test envelope. The pitch-up was accompanied by further load factor increase. Subsequently, a pronounced variation in the main rotor blade torsion trace on the oscillograph occurred because of a change in blade aerodynamic moment. This condition was referred to as blade moment stall. During pilot training for APE I.1, this characteristic was demonstrated at approximately 115 knots indicated airspeed (KIAS), 1.8g, and 60 degrees of bank angle. During this demonstration, a pronounced increase in 4/rev vibration occurred just prior to the nose-up pitch and rapid right roll associated with blade moment stall. The pitch and roll were sufficient to level the wings in a nose-high attitude from the 60-degree bank even though some left forward cyclic was applied to oppose it. During APE I.1 testing, deep blade moment stall was not encountered. However, the pronounced increase in 4/rev vibration prior to blade moment stall was encountered during left windup turns. For APE I.1 and the initial part of APE I.2, the safety-of-flight release envelope was limited to 1.5g in right windup turns because the contractor had not investigated blade moment stall to the right. During APE I.2, the contractor conducted turns to the right to approximately 2.0g, and the APE I.2 safety-of-flight release was amended to include 2.0g. An investigation of the characteristics of blade moment stall to the right has still not been conducted by the contractor.

It is recommended that additional testing be conducted to evaluate the blade moment stall characteristics up to the structural or control limits.

59. During APE I.3, blade moment stall was encountered. Figures 20 and 21, appendix H, are time histories of two encounters with blade moment stall during APE I.3A. Figure 18, appendix I, is a time history of blade moment stall occurrence during APE I.3B, which was the most severe stall encountered during APE and RDAT testing. While attempting to stabilize at 1.8g at 150 KCAS in a left windup turn, an uncommanded nose-up pitch was encountered. As the load factor began to increase, the pilot returned the cyclic to trim, then applied increasing forward longitudinal cyclic stick as the load factor continued to increase. The normal load factor reached approximately 2.55 (0.75g above target and 0.67g above the envelope limit) even though the longitudinal control was moved to the forward stop. During this left descending turn, the aircraft pitched up approximately 50 degrees and rolled to the right approximately 100 degrees. During the recovery, the collective blade angle was reduced to 3 degrees, and a minimum load factor of approximately 0.1 was reached (ref 23, app A). There was no timely warning apparent to the pilot preceding the uncommanded pitch-up. A slight increase of vibration was apparent at approximately 1.4g, but there was no perceptible increase up to the stall. The lack of satisfactory warning of uncommanded pitch-up is a shortcoming.

60. Pitch rate is a primary pilot cue during maneuvering flight. The current envelope load factor limits of the AH-56A are equivalent to steady pitch rates of approximately 6 deg/sec up to 150 KCAS decreasing to only 2 deg/sec at 200 KCAS. Excessive pilot attention is required to remain within the flight envelope. The present flight limitations and poor maneuvering characteristics of the AH-56A make it impractical to perform operational maneuvers at high speed. This inability to perform operational maneuvers at high speed is a shortcoming.

61. At low airspeeds (80 KCAS), maneuvering characteristics are satisfactory within the presently approved envelope. As airspeed increases, the weak-to-neutral stick-fixed and stick-free maneuvering stability make the longitudinal control very sensitive. This sensitivity is incompatible with the large control inputs required to initiate maneuvers. It was extremely difficult to stabilize on a load factor above 1.5 at airspeeds of 150 knots and above. Under these conditions, aircraft control is further complicated by the disharmony of the lateral and longitudinal control displacements and forces, by apparent roll oscillations which are a side effect of the lift/roll decoupler, and by pitch-due-to-sideslip coupling. At 150 KCAS and above, attempts to control load factor above 1.5 required excessive pilot attention inside the cockpit. Precise control of the aircraft should not require unique techniques at different airspeeds and load factors. During tactical maneuvering, the pilot's attention must be divided between many tasks. Although total pilot attention was devoted to aircraft control during maneuvering, control was lost, the load factor envelope was exceeded, and blade moment stall was encountered, on several occasions (HQRS 10). This inability to maintain aircraft control is a deficiency. Until this deficiency is corrected, the loss of control within the present flight envelope warrants a reduction in the envelope for future Army tests.

Controllability

General:

62. Controllability tests were conducted to evaluate the control power, response, and sensitivity as well as cross-coupling characteristics of the aircraft. Primary axis controllability was measured in terms of aircraft attitude displacements, angular velocities, and angular accelerations about an aircraft axis following a rapid step control input (maximum input time of 0.2 second) of a measured size. This input was held and all other controls held fixed until either the maximum rate was reached or recovery action was necessary. The magnitude of the inputs was varied (usually a minimum of three inputs in one direction for each control axis and one in the opposite direction). The inputs were started from a static trim condition in hover, level flight, and dives above V_H using an adjustable rigid control fixture to assist in achieving the desired inputs. Cross-coupling characteristics were evaluated by observing the angular displacements, velocities, and accelerations about all axes as well as cg normal acceleration and angle-of-sideslip changes which resulted from the step inputs. The data from these controllability tests are presented in figures 15 through 72, appendix F, and figures 19 through 33, appendix I. The data are summarized in terms of control response (maximum angular velocity per inch of stick displacement) and control sensitivity (maximum angular acceleration per inch of stick displacement) versus airspeed. Controllability testing was conducted during APE I.1 and APE I.3B at the conditions listed in table 4.

Longitudinal Controllability:

63. During APE I.1, the longitudinal control response to both forward and aft inputs was a constant 6 degrees per second per inch (deg/sec/in.) between airspeeds of about 60 and 100 KCAS (figs. 17 and 18, app F). At airspeeds less than 60 KCAS, the response to forward inputs increased with decreasing airspeed to a maximum of 8 deg/sec/in. at a hover. The response to aft inputs decreased in the same airspeed range to 5 deg/sec/in. at a hover. At airspeeds greater than 100 KCAS, response to forward and aft inputs increased with airspeed to maxima of 10 deg/sec/in. (aft) and 7 deg/sec/in. (forward) at 149 KCAS. The longitudinal control response is within the limits of CP0001A at airspeeds between hover and 149 KCAS. Longitudinal controllability tests were not conducted at airspeeds greater than 150 KCAS because load factor changes resulting from small control inputs would cause the aircraft to exceed the maneuvering flight envelope. Although quantitative longitudinal controllability data were not obtained at the higher airspeeds, the control response was qualitatively evaluated as excessive at airspeeds greater than 170 KCAS.

64. During APE I.3, the pilots reported that the longitudinal control felt "sluggish" at high speeds. This was evidenced by relatively large longitudinal control displacements required to initiate or stop a pitch rate when making small pitch attitude changes. To make a small pitch attitude change, the pilot initially made a small longitudinal control input. Following this input, time required for the pitch rate to increase to a perceptible level was sufficiently long that the pilot thought

the aircraft was not responding to his command. Therefore, the pilot applied a larger input and finally achieved a pitch rate. When approaching the desired pitch attitude, the pilot attempted to stop the pitch rate, but the aircraft overshot the desired attitude. The sluggishness at high speeds appeared to be caused by the long pitch time constant (time from initiation of control input to attainment of maximum rate). At 150 KCAS, the pitch time constant during APE I.3B was approximately 1.7 seconds as compared with 1.2 seconds during APE I.1. The difference was probably caused by the pitch desensitizer installed prior to APE I.3 (para 11c). This longer pitch time constant at high speeds degraded the ability to precisely control pitch attitude and complicated maneuvering tasks (para 61). The excessively long pitch time constant is a shortcoming.

65. Longitudinal control power is defined as the angular pitch displacement after 1 second following a 1-inch step control input. The data from APE I.1 testing are presented in figures 15 through 19, appendix F.

Table 4. Controllability Test Conditions.

Test	Approximate Calibrated Airspeed (kt)	Approximate Gross Weight (lb)	Longitudinal Control Axis	Lateral Control Axis	Directional Control Axis
APE I.1	Zero	18,300	X	X	X
	60	18,300	X	X	X
	100	18,300	X	X	X
	150	18,300	X	X	X
	180	18,300		X	
APE I.3B	Zero	18,300	X	X	X
	60	18,300	X	X	X
	100	18,300	X	X	X
	150	18,300	X	X	
	Zero	20,500	X	X	X
	150	20,500		X	

66. The APE I.1 longitudinal control sensitivity data are presented in figures 20 through 24, appendix F. The sensitivity was greater in the forward than in the aft direction at airspeeds from hover to 101 KCAS. Between 101 and 149 KCAS, this trend was reversed; and at 149 KCAS, the sensitivity in the aft direction was greater. Also at 149 KCAS, the sensitivity in the aft direction increased markedly from its value at 101 KCAS (the response had a similar characteristic). These test data correlated well with pilot comments that the longitudinal control was unacceptably sensitive at high speeds.

67. At 18,300 pounds, there appeared to be no significant difference between the longitudinal control response obtained during APE I.1 and that obtained during APE I.3B (figs. 19 through 22, app I); however, there was a significant difference between the longitudinal control sensitivity from the two tests. The APE I.3B data showed a reduction of 39 percent in sensitivity at 150 KCAS since APE I.1. At lower airspeeds, the sensitivity was unchanged. The change at 150 KCAS was apparently caused by the pitch desensitizer system installed prior to APE I.3 (para 11c). This system is effective only above 110 knots and was probably responsible for the slight reduction in response at 150 KCAS. With the pitch desensitizer system installed, the excessive longitudinal control sensitivity observed during APE I.1 was no longer objectionable.

Lateral Controllability:

68. The APE I.1 lateral control response (figs. 25 through 31, app F) increased slightly with increasing airspeed from 23.5 deg/sec/in. to a maximum of 27 deg/sec/in. at 149 KCAS. Above this airspeed, the response began decreasing with increasing airspeed to a value of 24 deg/sec/in. at 179 KCAS. At all airspeeds tested, the lateral response was above the 20-deg/sec/in. maximum limit of paragraph 3.3.15 of MIL-H-8501A, but response was not objectionable.

69. The lateral control power (roll attitude displacement after 1/2 second following a 1-inch lateral control step input) satisfied the requirement of MIL-H-8501A (fig. 26, app F). The control power exceeded the hover requirement at all airspeeds tested during APE I.1. This is a desirable characteristic for an attack aircraft.

70. The APE I.1 lateral control sensitivity (figs. 32 through 38, app F) increased with increasing airspeed from 53 deg/sec2/in. at a hover to 62 deg/sec2/in. at 179 KCAS, the maximum speed tested. The sensitivity increase between 60 and 165 KCAS was approximately linear and changed only about 2 deg/sec2/in.

71. The lateral control response and sensitivity were greatly reduced following APE I.3A as part of a program to eliminate the tendency toward pilot-coupled roll oscillations (para 11d). At a nominal gross weight of 18,300 pounds, the response obtained during APE I.3B was reduced to 7 deg/sec/in. at a hover and to 17 deg/sec/in. at 149 KCAS (figs. 23 through 26, app I). This response meets the minimum requirement of 5 deg/sec/in. of CP0001A. The sensitivity was reduced to 19.5 deg/sec/in. at a hover, increasing with increasing airspeed to 36 deg/sec2/in.

at 149 KCAS. The lateral control response and sensitivity were less at 20,500 pounds (with external stores) than at 18,300 pounds (with no external stores). The roll response of the AH-56A was reported as an excellent feature during APE I.1, enhancing the agility of the aircraft. During APE I.3, the roll response was reduced significantly, degrading a desirable characteristic. Further testing is recommended to optimize the roll response for the attack mission.

72. The tendency toward pilot-coupled roll oscillation observed during APE I.1 was reduced prior to APE I.3B (paras 104 through 111). The reduction of this tendency was probably due more to the reduction of lateral sensitivity than to the reduction of lateral response. Other Army helicopters have lateral response similar to the Cheyenne's response during APE I.1, but pilots do not have the tendency to couple in roll. The control sensitivity of those aircraft, however, is less than half the sensitivity of the Cheyenne during APE I.1. Although reduction of the Cheyenne's lateral control sensitivity may have been necessary to reduce coupling tendencies, the reduction of lateral response degraded an excellent feature of the aircraft.

Directional Controllability:

73. The directional controllability data obtained during APE I.1 are presented in figures 39 through 48, appendix F. These data were obtained without the use of a control fixture. The results of these data show that control response to both left and right pedal inputs ranged from 12 to 17 deg/sec/in. between airspeeds of 62 and 150 KCAS. The maximum rates were reached within 1 second of initiation of control input at these airspeeds. Response in a hover is presented in terms of yaw rate at 1 second after the pedal input versus the size of the input because the yaw rate continued to increase without reaching a steady value. Directional controllability tests were not conducted at airspeeds greater than 150 KCAS so that the small sideslip and maneuvering limits of the safety-of-flight release would not be exceeded. Directional control power could not be evaluated because of unreliable yaw attitude instrumentation.

74. From the APE I.1 data, the directional sensitivity (figs. 44 through 48, app F) appeared to be above 30 deg/sec2/in. at all conditions tested, except to the right at a hover and in both directions at 62 KCAS. Above 62 KCAS, the directional sensitivity increases with increased airspeed. The directional control response and sensitivity were satisfactory.

75. The data from APE I.3B directional controllability tests (figs. 27 through 29, app I) indicate no significant change in response from that obtained during APE I.1. The sensitivity obtained during APE I.3B, however, is lower than APE I.1 sensitivity, particularly at 100 KCAS. This difference is believed to be caused by the lack of a control fixture during APE I.1. Because of this, the pilot sometimes overshot the desired input size initially and then returned to the desired size, which would produce larger angular accelerations without necessarily altering the response data. The directional control response and sensitivity were satisfactory for the conditions tested.

Cross-Coupling

General:

76. During the conduct of the controllability tests, significant cross-coupling was observed, particularly during APE I.1. The three general types found were pitch due to sideslip, roll due to lift, and pitch due to roll. Pitch due to sideslip also appeared as a longitudinal trim change with sideslip angle in the static lateral-directional tests (para 44). Roll due to lift also appeared as a lateral stick displacement with load factor during the maneuvering stability tests (para 54).

Pitch Due to Sideslip:

77. During APE I.1, pedal step inputs in a hover produced negligible pitch and roll rates (fig. 66, app F). In forward flight, however, the same type inputs produced significant pitch rates and associated load factor and roll rate changes. A left pedal input produced a nose-up pitch rate and increase in load factor, and a right pedal input produced a nose-down pitch rate and a reduction in load factor. Figures 67 through 71 show pedal step inputs of approximately 1/2 inch at nominal airspeeds of 60, 100, and 150 KCAS. The maximum sideslip angles achieved decreased with increasing airspeeds, but the maximum pitch rates remained essentially constant. This means that the pitch rate per degree of sideslip angle increased with increasing airspeed. This is consistent with the trends of static lateral-directional stability with airspeed which are presented in figures 6 through 9. The maximum load factor changes increased with increasing airspeed as expected, even though the pitch rates were constant. Because of the roll-due-to-lift coupling in this aircraft (para 54), load factor excursions also produced roll rates (left roll with decreasing load factor and right roll with increasing load factor). The roll rates encountered during these tests were not considered objectionable by the pilot. The pitch due to sideslip, however, was objectionable.

78. The pitch due to sideslip encountered during APE I.3B tests was essentially the same as encountered during APE I.1. This pitch due to sideslip did not produce uncontrollable pitch rates but was very annoying. Quickly changing aircraft heading (*ie*. targets) in a dive was particularly difficult since some sideslip was almost always introduced requiring pilot compensation. This coupling also adversely affected maneuvering characteristics (para 57). The pitch-due-to-sideslip coupling is a shortcoming.

Roll Due to Lift (Lift/Roll Coupling):

79. During hover longitudinal controllability tests, lift/roll coupling was not apparent since there was no change of load factor (fig. 49, app F). In addition, there was negligible roll due to pitch. In forward flight during APE I.1, however, lift/roll coupling was quite apparent. It was manifested as right roll rate with load factors greater than 1.0 and left roll rate with load factors less than 1.0 (figs. 50 through 54). At a constant airspeed, the roll rate varied with load factor, as would be expected from the maneuvering stability tests. Good agreement is shown by

a comparison of the roll rates obtained during longitudinal controllability with the lateral stick displacement with load factor found during the maneuvering stability tests (para 54). For example, the maximum roll rate achieved following an aft longitudinal step input at about 150 KCAS (fig. 53, app F) was slightly more than 20 deg/sec (when corrected for a small lateral cyclic input), and the maximum load factor was 1.6. The lateral stick displacement in a left windup turn was 0.8 inch at 1.6g (fig. 12), which is equivalent to a 21.6-deg/sec roll rate (at 1g) from a control response standpoint (fig. 25). Since the pitch rate during these two maneuvers was significantly different and the roll moments were equal, roll-due-to-pitch coupling was not present.

80. The variation of roll rate with airspeed at a constant load factor appeared to follow the trend of lateral stick displacement determined in the maneuvering stability tests (fig. 10, app F). The lift/roll coupling was most severe in the 145 to 165 KCAS range, with the least severe coupling below 100 KCAS. Although the best means of quantifying lift/roll coupling is in terms of lateral stick displacement per g (obtained under steady-state conditions during windup turns), the effects of this coupling on pilot workload and the ability of the pilot to control the aircraft are best demonstrated in dynamic maneuvers. In this respect, the recovery from an aft longitudinal step at 148 KCAS (fig. 53) is of interest. Prior to recovery, the pilot sensed the right roll caused by lift/roll coupling in addition to the nose-up pitch rate and the increased load factor. Therefore, upon recovery, the pilot displaced the cyclic control left as well as forward. Since the right rolling moment was reduced by the reduction of load factor, the left lateral displacement of the cyclic produced a left roll rate larger than desired by the pilot. At this point, the pilot entered pilot-coupled roll oscillations (paras 104 through 111) for approximately 2 cycles. The pilot reported difficulty in maintaining a wings-level attitude during symmetrical pull-ups and pushovers and also a tendency to overcontrol in roll, particularly during pushovers.

81. The lift/roll coupling in the AH-56A was significantly reduced by the lift/roll decoupler installed prior to APE I.3B (para 11d). The decoupler senses load factor and airspeed and makes control inputs through a modulation piston in the roll servo. The roll moments produced by these inputs oppose the moment created by lift/roll coupling. The rolling moments did not always cancel each other because of the phasing and gain of the decoupler, and the residual lift/roll coupling was still objectionable. As was determined during APE I.1, the coupling was most severe at approxmately 150 KCAS. These characteristics were apparent during longitudinal controllability testing during APE I.3B. Longitudinal step inputs at 150 KCAS (figs. 30 and 31, app I), produced an apparent roll oscillation and a residual roll rate. This characteristic caused difficulty in precisely controlling roll attitude while maneuvering (paras 55 and 56). The lift/roll coupling and uncommanded roll oscillations which occurred whenever the normal acceleration of the aircraft was changing are shortcomings.

Pitch Due to Roll (Pitch/Roll Coupling):

82. In APE 1.1, during lateral controllability tests in a hover, pitch/roll coupling was not apparent (figs. 55 and 56, app F) but was apparent in forward flight. Large right lateral step inputs (up to 1.6 inches) at nominal airspeeds of 150, 165, and 180 KCAS (figs. 60, 62, and 64, respectively) produced excessive pitch/roll coupling. Right lateral steps at these speeds produced a slight initial nose-up pitch rate followed by a large nose-down pitch rate and associated reduction in load factor. The initial nose-up pitch rate was not noticeable to the pilot. The large reduction in load factor caused the pilot to initiate recovery. On some occasions during the recovery, the load factor reached a minimum value of 0.4. The pitch/roll coupling at 150, 165, and 180 KCAS became more severe with larger lateral inputs (*ie*, larger right roll rates). Left lateral step inputs at these speeds produced a very slight initial nose-down pitch rate followed by a larger nose-up pitch rate (figs. 61, 63, and 65). The magnitude of the nose-up pitch rate increased with increasing left roll rate, but in all cases was less severe than the pitch down with comparable right roll rates. In left rolls, an increase in 4/rev vibration occurred. For small control inputs in both directions, the pitch/roll coupling was not objectionable. At the lower airspeeds tested, the pitch/roll coupling was less severe and sometimes masked by the pitch due to sideslip. During right lateral steps at 63 and 100 KCAS (figs. 57 and 59), the nose pitched up rather than down, apparently because of the large right sideslip introduced during these maneuvers. Left lateral steps at 63 and 100 KCAS produced little or no pitch rate. In general, the pitch/roll coupling was more severe with right lateral inputs (nose-down pitch). The coupling was negligible in a hover and increased in severity with increasing airspeed, reaching an unacceptable level at about 150 KCAS. The pitch rates caused by pitch/roll coupling were higher with higher roll rates. The pitch/roll coupling encountered during APE 1.1 degraded the aircraft suitability as an attack helicopter.

83. Prior to APE 1.3, a control augmentation system was installed which virtually eliminated pitch/roll coupling (para 11c). During initial RDAT I testing, however, aircraft pitching with high roll accelerations was encountered with this system installed. The pilot described this as "nose bobble" with easily commanded high roll accelerations. Subsequent to the initial RDAT I testing and prior to the lateral controllability testing of APE 1.3B, the lateral control sensitivity was reduced markedly (para 11d). With this reduction in sensitivity, the roll accelerations were less during APE 1.3B than during RDAT I. As a result, the "nose bobble" encountered during the RDAT I testing was not apparent during APE 1.3B. Aircraft reactions following a typical lateral step during APE 1.3B are presented in figure 32, appendix I. Pitch/roll coupling is no longer a problem in the AH-56A.

Interaction of Cross-Coupling and Effects on Primary Axis Response:

84. The three types of cross-coupling found during APE 1.1 (pitch due to sideslip, roll due to lift, and pitch due to roll) did not always appear independently. There was often interaction between them, and under some conditions, this undesirable interaction affected the primary response of the aircraft. For example, at the higher

airspeed, large lateral inputs caused roll rates which peaked rapidly. By the time the roll rate reached a maximum, however, pitch/roll coupling had produced significant pitch rates and attendant load factor changes. Because of lift/roll coupling, these load factor changes produced roll moments which opposed the moment created by the lateral control input. Therefore, the roll rate decreased rapidly, even though the control input was held constant (figs. 60 through 65, app F). This interaction between pitch/roll and lift/roll coupling aggravated the tendency for the pilot to overcontrol in roll at high speed, as is evidenced in figure 53, and to a lesser degree in figures 59 through 64. The pilot sensed the maximum roll rate and reacted to it by applying lateral cyclic in the opposite direction at about the same time the cross-coupling was also reducing these rates.

85. The magnitudes of the three types of coupling varied with both airspeed and magnitude of control input. Additionally, the polarity of pitch coupling with lateral control inputs changes with airspeed. These facts, plus the complex interaction of the various types of coupling, indicated that even with more experience in the aircraft, a pilot could not adequately compensate for uncommanded aircraft responses, even though the pilot's full attention was directed to that end. Control of the aircraft was further degraded when pilot attention was divided between flying the aircraft and other mission tasks.

86. During APE 1.3B, the effect of cross-coupling on primary axis response was negligible because of the elimination of pitch/roll coupling. However, since lift/roll and pitch due to sideslip were still present, the pilot had considerable difficulty compensating for uncommanded responses.

Dynamic Stability

87. Dynamic stability tests were conducted to evaluate the aircraft short-period response following a gust disturbance. Gust disturbances were simulated by making 1-inch pulse control inputs, which were held for 0.5 second. Following the input, all controls were held fixed until either the aircraft motions damped or recovery action was required. Dynamic stability characteristics were evaluated in both directions for longitudinal, lateral, and directional controls. These tests were conducted during APE I.1 and APE 1.3B at the same conditions as the controllability testing (table 4).

88. Longitudinal disturbances were well damped (figs. 73 through 75, app F). The short-period longitudinal dynamic stability characteristics of the AH-56A were satisfactory for all conditions tested during APE 1.

89. During APE 1.1, the lateral dynamic stability characteristics of the aircraft were investigated. Following a pulse control input, a time history of the roll rate showed a small amplitude roll oscillation which was lightly damped (figs. 76 through 80, app F). The initial period of this oscillation was approximately 0.7 second per cycle at airspeeds from hover to 100 KCAS, decreasing to approximately 0.6 second at 180 KCAS. At airspeeds below approximately 64 KCAS, the damping ratio of this oscillation (approximately 0.1 at 64 KCAS

in level flight and 0.03 in a hover) was below the minimum requirement of paragraph 3.6.1.2 of MIL-H-8501A. At airspeeds of 100, 150, and 180 KCAS, the period appeared to increase with each cycle. This increase was greatest at 150 KCAS and was very slight at 180 KCAS. The initial damping appeared to be at or below the minimum required by the military specification at airspeeds above 65 KCAS. The damping appeared to increase with each cycle at airspeeds of 100, 150, and 180 KCAS. The lateral dynamic stability characteristics of the aircraft were unsatisfactory during APE I.1.

90. Roll oscillations contributed to the tendency of the pilot to couple with the aircraft in roll throughout the flight envelope. At the higher airspeeds (above 100 KCAS), this tendency was not excessive; however, it caused an increase in the pilot workload. At the lower airspeeds (below 65 KCAS), the coupling tendency was significantly increased. This was especially true in the transition portion of the flight and during liftoff to and touchdown from a hover. Under these conditions, large lateral trim changes occurred which excited the roll oscillation. Also, at these lower airspeeds, the damping of the roll oscillation was very weak. The large lateral trim changes, the weak lateral damping, and the high lateral control response and sensitivity often resulted in pilot-coupled roll oscillations (fig. 85, app F). A further discussion of pilot-coupled roll oscillations can be found in paragraphs 104 through 111.

91. The roll oscillation discussed in paragraph 89 has been attributed to the response of the coupled inplane-roll mode. In this context, this roll oscillation is of more concern, in that it not only increases the pilot workload, as discussed in paragraph 90, but it is also associated with the structural dynamics of the main rotor. A deterioration of the damping of this oscillation would cause it to become more pronounced and perhaps unstable. This could cause a dynamic instability in the main rotor. During APE I.3A, the contractor conducted several control free-play checks in an attempt to minimize the effect of roll oscillations. The effect of component deterioration (free play, friction, damping, etc.) on the damping of this oscillation is not known. A detailed investigation should be made to determine the effect and the limits of deterioration.

92. The lateral short-period dynamic stability of the aircraft was greatly improved prior to APE I.3B. A roll stability augmentation system (called a roll compensator by the contractor) is responsible for this improvement (para 11d). Testing during APE I.3B has shown that lateral disturbances are well damped (figs. 34 and 35, app I). The contractor accomplished extensive testing in the low-airspeed regime with the roll compensator OFF between APE I.1 and APE I.3B. The results of some of those tests are presented in reference 24, appendix A, and indicate that the roll damping with the roll compensator OFF has decreased since APE I.1 and is unacceptably low at airspeeds less than approximately 50 KCAS. It is believed that this decrease in aircraft roll damping is caused by the decreased damping of a main rotor oscillatory mode (regressive mode). The regressive mode damping decreased as a result of increased main rotor droop. Damping of the characteristic 1-Hz roll oscillation was minimum in the 25- to 35-knot airspeed range during the contractor testing. This is very near the airspeed at which the lateral trim

shift is encountered when transitioning from hover to forward flight. This trim shift and low roll damping contributed to the problem of pilot-coupled roll oscillations encountered during APE I.1 and APE I.3A (paras 104 through 111). Hover takeoffs and landings should not be attempted with the roll compensator inoperative. With the roll compensator ON, the roll damping is satisfactory.

93. Directional disturbances were well damped (figs. 81 through 84, app F). The short-period directional dynamic stability characteristics were satisfactory throughout the test program.

94. During this program, specific testing to evaluate the 1P X 2P instability was not conducted. However, during a takeoff early in APE I.1, inadvertent rotor speed operation down to 83-percent rpm and high shaft moments were encountered. Low rotor speed and high shaft moments are prerequisites for 1P X 2P instability; however, this instability was not apparent.

95. During this program, specific testing to evaluate the 0.55P "hop" instability was not conducted. This instability was not apparent.

Takeoff and Landing Characterisitcs

Hover:

96. The pilot effort required to hover the AH-56A in prepared areas was not excessive. Liftoff to a hover resulted in a change of roll attitude from 3 degrees (right wing low) on the ground to 5 degrees (left wing low) in a hover. The natural tendency was to attempt to hold the aircraft level using lateral cyclic control which often resulted in overcontrolling. Normal helicopter techniques were used to hover with the addition of propeller pitch control. Propeller pitch was varied, depending on wind conditions, to maintain a fixed position over the ground. The hover detent which requires minimum power under no-wind conditions is -2.2 degrees; this setting produced a level pitch attitude. Increases of propeller pitch resulted in a forward movement which was corrected by aft control displacemer producing a nose-up pitch attitude. The ability to stabilize in various pitch : .tudes in a hover is a unique and desirable feature of the Cheyenne. The aircraft characteristically hovered in a left-wing-low attitude (approximately 5 degrees) which was not objectionable to the pilots, but fails to meet the requirements of CP0001A which specifies a level hover attitude.

Hover Takeoff:

97. Transition to forward flight from a hover was initiated by slowly increasing propeller thrust while maintaining attitude with the cyclic and directional controls and height with the collective. During acceleration through translation, pronounced trim changes occurred in all axes. The aircraft tended to roll right, pitch up, and yaw left, requiring large control corrections and trim changes. The size of the change depended on the gross weight. The rate of change depended on the forward acceleration. The lateral control change was approximately 1-1/2 inches; the

longitudinal change was approximately 2 inches; and the directional change was approximately 2 inches (approximately 30 pounds including breakout). Overcontrolling was observed in all axes and was particularly objectionable in the roll axis while correcting for roll perturbations resulting from the transition trim shift, gust disturbances, collective and propeller pitch changes, or any other disturbance. Prior to APE I.3B, overcontrolling led to pilot coupling which is discussed in paragraphs 104 through 111. This coupling did not occur during APE I.3B testing. Pitch and roll trim changes were not objectionable, but the directional trim change was objectionable because of the large magnitude and because of the functioning of the directional trim system. The directional trim system is discussed in paragraph 29. Power management was complicated by the need to correlate two power controls (propeller pitch and main rotor collective pitch) and is further discussed in paragraph 112.

Hover Landing:

98. Returning to a hover from forward flight presented the same magnitude changes of control requirements in the opposite direction. The same tendency to overcontrol in roll observed during hover takeoffs was present. The directional trim characteristics (para 29) caused out-of-trim conditions which often resulted in large heading variations.

Unimproved Area Operations:

99. During RDAT I, hover takeoffs and landings were made into unimproved firing positions in the desert conditions characteristic of Yuma Proving Ground, Arizona. Approaching and establishing a high-hover, and transitioning from the hover to forward flight over the unprepared firing position were difficult and demanding flight tasks. The pilot tasks and workload involved in making an approach to the hover over uneven terrain to an unprepared point were greater in this compound helicopter than the comparable task in a pure helicopter. Varying forward thrust using the propeller greatly increased the spectrum of approach variables that the pilot had to control. A natural tendency is to establish a hover with reference to the terrain. This tendency may result in a nose-high or nose-low attitude which requires more power to hover because of the additional propeller thrust. The pilot workload during the test was increased because the conditions of temperature, gross weight, and hover height resulted in a very small margin between power required and the maximum power available. Pilot training must emphasize the characteristics of varying propeller thrust and its effect on safe operation to and from unimproved areas.

100. Acceleration to forward flight was made using the level-attitude technique. The propeller pitch was increased until 99-percent power was reached. The resulting acceleration was slow and a dust cloud developed which completely enveloped the aircraft for several seconds. Recorded roll oscillations during the acceleration were very low (less than ±3 deg/sec). During the approach to the hover, roll rates of up to ±6 deg/sec were recorded. No control difficulties or aircraft roll oscillations were detected by the pilot or copilot/gunner during either the approach or acceleration maneuvers.

Rolling Takeoff:

101. Rolling takeoffs were easier to accomplish than hovering takeoffs. The method used was to accelerate the aircraft to liftoff speed (50 to 60 KIAS) using propeller thrust, with the collective at 3 degrees, and then to make a small smooth collective application to become airborne. Large nose-up trim shifts resulted from the collective application which had to be corrected to keep the aircraft in a level accelerating attitude. Compared with the hovering takeoffs, there was less tendency to couple with the aircraft in roll, power management was not as critical (para 112), and the vibration levels were lower.

Run-On Landings:

102. Run-on landings required less pilot attention and were generally more smoothly accomplished than hover landings. Aircraft attitude control was less critical than during hover landings because the aircraft was landed at a speed above the speed at which reduced roll damping was encountered. In addition, power management was less critical because the high power requirements of hover were avoided. The task did become more difficult when collective pitch was increased too rapidly resulting in a large roll trim shift just prior to touchdown. A time history of this trim shift and the resulting roll oscillation during APE I.1 is presented in figure 86, appendix F. In this case, the rolling landing was aborted, and the approach was terminated at a hover due to the roll oscillations.

103. The landing gear struts failed to compress evenly during several landings. Frequently, when run-on landings were made, the right strut compressed first resulting in a right roll as collective was lowered. The left strut would then compress. On one occasion, the roll rate was so abrupt that the landing was aborted to a hover to allow a verification of gear extension. The left strut was found to hang up during hover landings when the collective was slowly lowered so that smooth ground contact was made. With the collective pitch at the pneumatic-down stop (3.2 degrees), the roll attitude was 5 degrees (right wing down) when the left strut failed to compress (ref 25, app A). Failure of the two main landing gear struts to compress evenly complicates the landing task, and is a shortcoming.

Lateral Overcontrolling and Pilot-Coupled Roll Oscillations

104. Lateral overcontrolling was observed during APE I.1, APE I.2, RDAT I, and APE I.3. Pilot-coupled roll oscillations resulted from overcontrolling in Cheyenne 66-8834 (LCC SN 1009) although pilot-coupled roll oscillation was not reported in Cheyenne 66-8831 (LCC SN 1006). A development program was conducted by Lockheed-California Company to attempt to eliminate overcontrolling and pilot-coupled roll oscillation. This program and the results of USAASTA tests which were conducted are discussed in the following paragraphs. The aircraft characteristics observed during APE I.3B are also discussed.

105. Overcontrolling is defined as the use of more control displacement than is necessary to return the aircraft to the desired attitude following a disturbance,

requiring that the pilot make an additional correction to attain the desired aircraft attitude. Pilot coupling is a specific case of overcontrolling in which the phasing of the pilot's control input reinforces rather than corrects the aircraft oscillations. The pilot's response time is the time required to recognize the aircraft disturbance, decide on the corrective action, and move the control. Pilot response times normally range from 0.5 to 0.8 second. Aircraft response is determined by the control system lag and the short-period response. When the aircraft response has a period in the range of 0.8 to 1.2 seconds, coupling may occur when an overcontrolling tendency is present, as is the case in the AH-56A.

106. During APE I.1, overcontrolling or pilot-coupled roll oscillations were present during each hover landing and hover takeoff. The tendency to overcontrol while correcting for roll disturbances has also been observed throughout the flight envelope during APE and RDAT testing. Pilot-coupled roll oscillation was also a signficant factor in complicating recovery from blade moment stall (refs 26 and 27). Pilot-coupled roll oscillations during hover takeoffs and landings were a serious deficiency during APE I.1 (ref 15) and APE I.3A, and the tendency to overcontrol in roll was encountered during every task evaluated in RDAT I and was reported as a deficiency (ref 17). Hover takeoffs and landings were precluded during APE I.2 due to limitations of the allowable collective blade angle. Time histories of hover takeoffs, landings, recoveries from blade moment stall, and recovery from a lateral step are presented in figures 85 through 88, appendix F, and figures 20 through 24, appendix H. Pilot-coupled roll oscillations observed while flying the AH-56A were found to have two particularly serious results: (1) aircraft control requirements increased the pilot workload to such a degree that the pilot was unable to cope with other critical functions such as power management; and (2) at low airspeeds, main rotor chord bending loads increased significantly.

107. The increase of workload imposed by overcontroling and pilot-coupled roll oscillation was most severe during hover takeoffs and landings where the pilot workload is at the highest level observed in the flight envelope. These tasks are much more complex in the Cheyenne than in other aircraft in the Army inventory due to the additional control requirements of the propeller and the large control position trim changes which occur in all axes when passing through transition (paras 97, 98, and 112). In addition, roll damping decreases to a minimum at approximately 35 KIAS (para 92). On numerous takeoffs and landings, intense pilot attention was required to control the aircraft while attempting to recover from the pilot-coupled roll oscillation (ref 28, app A). During one instance, power management was neglected and the main rotor speed dropped to 83 percent due to power demands in excess of the engine capability. During rolling takeoffs or landings, the pilot workload was less since the large lateral and directional trim changes and minimum damping occurred when the aircraft was on the ground. Additionally, power management was not as critical.

108. Large increases in main rotor loads occur during pilot-coupled roll oscillation in the AH-56A. During one transition, loads approached 98 percent of the buckling limit load. During the instances of coupling shown in figures 22 and 23,

appendix H, the main rotor chord bending loads exceeded the inspection limit by approximately 25 percent and 10 percent, respectively (refs 29 and 30, app A); however, no apparent damage was revealed during inspection of the blades.

109. The Lockheed-California Company introduced a series of changes in an attempt to eliminate pilot-coupled roll oscillation. These changes included: installation of the roll compensator, reduction of control system free play, reduction of lateral control breakout and force gradient, installation of 16-Hz and 32-Hz notch filters in the roll compensator, reduction of lateral control sensitivity and response, and removal of the main rotor tip weights. Although removal of the tip weights was effective in increasing roll damping, the weights were reinstalled in order to maintain sufficient damping of a main rotor oscillatory mode (reactionless mode) (ref 31, app A). The chronology of testing and modifications associated with pilot-coupled roll oscillation is presented in figure A.

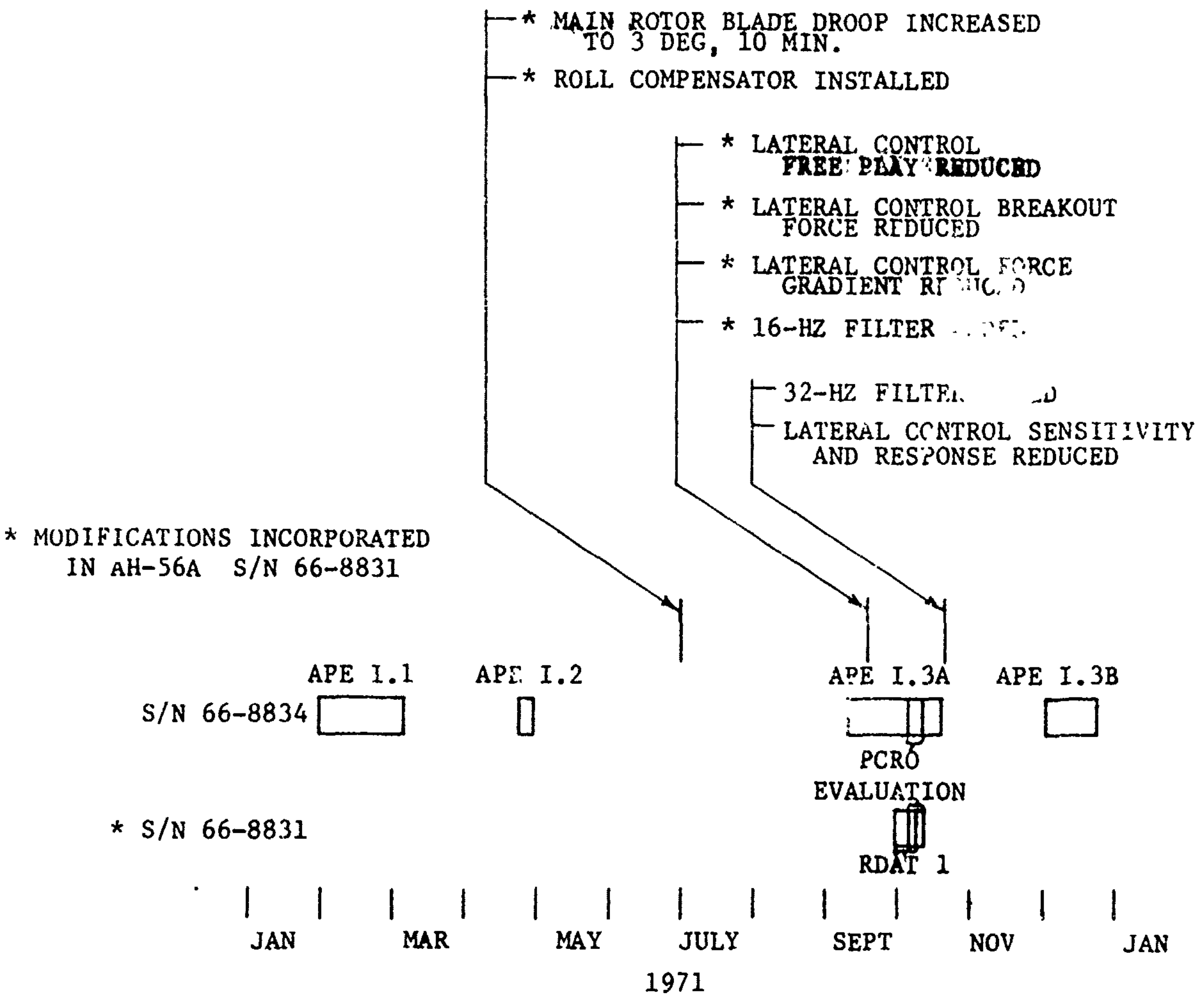

Figure A. Relation of Pilot-Coupled Roll Oscillation Modifications to APE I, RDAT I, and Pilot-Coupled Roll Oscillation Testing.

110. Evaluations of the initial and second group of modifications were conducted during APE I.3A (fig. A). An additional comprehensive evaluation was directed. The results of this evaluation (ref 32, app A) confirmed the APE I.3A results and are as follows:

a. The installation of the roll compensator increased roll damping in the low-airspeed regime, except around 35 KTAS, but did not prevent pilot-coupled roll oscillation.

b. The 16-Hz notch filter was required to prevent saturation of the roll compensator due to high vibration.

c. Changes to the lateral control breakout and force gradient increased the tendency to overcontrol at higher airspeeds (above 100 knots).

d. The various techniques used during hover takeoff provided no significant differences in the tendency toward pilot-coupled roll oscillation.

e. There was a greater tendency toward pilot-coupled roll oscillation noted when flying from the forward cockpit.

f. Increasing gross weight from 18,300 to 20,500 pounds resulted in an increase of vibration, produced larger trim shifts, and increased pilot attention required for power management, all of which increased the tendency towards pilot-coupled roll oscillation.

g. No siginficant differences were found between the handling qualities of aircraft S/N 66-8831 and S/N 66-8834.

h. Diversion of attention from aircraft control during critical phases of takeoff and landing was immediately followed by pilot-coupled roll oscillation.

111. Prior initiation of APE I.3B, the final modifications of this series were incorporated: reduction of the lateral control sensitivity and response and installation of the 32-Hz filter in the roll compensator. The aircraft handling qualities were evaluated during a total of 18 hover takeoffs and landings by two USAASTA pilots. During these transitions, no pilot-coupled roll oscillations were encountered, although roll overcontrolling was observed. During hover takeoffs at 20,500 pounds, pilots reported an apparent reduction of roll damping at approximately 35 knots giving a feeling similar to hover takeoffs at 18,300 pounds prior to the installation of the roll compensator. The tendency toward pilot-coupled roll oscillation has been significantly reduced, but the tendency to overcontrol in roll is still present, particularly during hover takeoffs and landings (HQRS 4). The overcontrolling tendency fails to meet the requirements of paragraph 3.3.15 of MIL-H-8501A and is a shortcoming.

Power Management

112. Power management required higher pilot workload in this aircraft than in conventional helicopters because of the additional control requirement of the pusher propeller. Pilot workload was high, but acceptable, during takeoffs from a hover. During hover takeoffs, the pilot accelerated the aircraft by increasing propeller blade angle. When translational lift was achieved, the pilot began reducing the collective blade angle while continuing to increase propeller blade angle until the desired settings for climb were reached. The pilot had to carefully program these changes in propeller and collective blade angles to avoid exceeding maximum power limits. This was especially critical during the first part of the takeoff when the relatively high power required to hover left only a small amount of power for the propeller. This problem will become more critical as gross weight, altitude, or temperature is increased. Timely warning of rotor speed loss because of power demands exceeding power available is required. During APE I.1, the warning did not activate until rotor speed had reached 90 percent, the lower limit. Prior to APE I.3, the warning was changed to activate at 96 percent, allowing the pilot to react and remain within limits.

113. During high-speed flight at high power settings, power required was very sensitive to airspeed changes at fixed propeller blade angles. Power required was also very sensitive to small propeller control changes. Therefore, the pilot had to closely monitor torque, turbine inlet temperature, and main rotor speed in this flight regime to avoid exceeding maximum power limits. It was difficult to know which parameter (torque or turbine inlet temperature) would determine maximum power available. A characteristic which complicated the pilot's task was the difficulty of differentiating between rpm loss caused by engine topping and that caused by static droop. Therefore, the pilot workload was unacceptably high in the high-speed flight regime and is a shortcoming. At all airspeeds, power increases with a decrease in airspeed at a constant propeller blade angle. Since this is opposite to power requirements in other aircraft, it must be stressed in AH-56A pilot training.

Autorotational Entry

114. Autorotational entries were performed during APE I.3B at 150 and 160 KCAS using a collective of 5 degrees and Beta for level flight, and at 180 KCAS using a collective of 3 degrees and Beta for normal rated power. Tests were initiated at 100-percent rotor speed and a density altitude of 4000 to 5000 feet. Takeoff gross weight was 18,800 pounds with the cg at FS 300.0 (aft) in the clean configuration. Engine failures were simulated by use of the engine overspeed protection system which restricted fuel flow when activated by a pushbutton on the cyclic stick. Change of the collective pitch or propeller pitch (Beta) control was progressively delayed in order to evaluate the aircraft response and determine the time available for recognition and reaction to a sudden, complete engine power failure (delay time).

115. Within the airspeed range of 150 to 180 KCAS, aircraft attitude excursions following simulated engine failures were mild. Initially, a left yaw occurred followed

by a right yaw resulting from reduction of propeller pitch. Pitch and roll excursions were minor until the collective control was lowered. At airspeeds above 150 KCAS, the primary rotor-speed control is Beta, and care should be exercised to avoid a rapid reduction of collective which can result in abrupt nose-down pitching and further rotor speed decay.

116. The decay rate of main rotor speed prior to reduction of collective pitch or propeller pitch varied between 10 and 14 percent per second at 180 KCAS. Automatic reduction of propeller pitch to 18 degrees was initiated by the Delta Beta system when main rotor speed decreased to 94 percent. During these tests. the maximum pilot delay achieved was 0.8 second, and the transient minimum rotor speed observed was 91 percent. Activation time of the Delta Beta system was consistently greater than 1.2 seconds following a simulated engine failure. This response time is too long to prevent the rotor speed from falling below the lower transient limit (85 percent) within the 2-second delay time required by MIL-H-8501A. When considering the many tasks that will occupy the pilot during attack missions, the excessive rate of rotor speed decay and delayed activation of the Delta Beta system will not allow the pilot to recognize the engine failure and take proper corrective action in sufficient time to keep within the operating limits. The excessive rotor speed decay at high airspeed which allows the rotor speed to drop below the minimum transient limits is a deficiency. A complete evaluation of the autorotational entry, descent, and landing characteristics should be conducted by the Army.

117. The phenomenon defined as wing stall was observed during several stabilized autorotational descents at approximately 85 KIAS. Wing stall produced a mild rolling oscillation similar to light turbulence. Following the stall, main rotor speed increased approximately 3 percent. Wing stall was not objectionable.

Pusher Propeller Characteristics

118. The pusher propeller is a unique feature of the AH-56A compound aircraft providing an excellent capability to make rapid decelerations, stabilize airspeed during dives, and permit a variation of aircraft attitudes in stablilized flight Aircraft control requirements with propeller pitch changes are easily controllable at airspeeds above 100 KIAS, but large and objectionable directional trim shifts result during landing approach and takeoffs. The capability to rapidly decelerate the aircraft enhances agility for nap-of-the-earth flight, and the capability to hold a constant airspeed during dives increases the time available for target engagement. Dive characteristics were evaluated at a trim speed of 120 KCAS with the collective pitch set at 5 degrees. A dive angle of 9 degrees and a rate of descent of 200 feet per minute resulted from a 16-degree propeller setting; a dive angle of 12 degrees and a rate of descent of 2400 feet per minute resulted from a 12-degree propeller setting; and a dive angle of 16 degrees and a rate of descent of 3500 feet per minute resulted from an 8-degree propeller setting.

Sideward and Rearward Flight

119. Sideward and rearward flight tests were conducted to simulate hovering in winds from zero to 20 KTAS, at azimuths of zero, 45, 90, 135, 180, 225, 270, and 315 degrees relative to the aircraft. The tests were conducted during APE 1.3B at a nominal gross weight of 20,500 pounds, a forward cg at FS 293, and a wheel height of approximately 5 feet. The data are presented in figures 37 through 40, appendix I.

120. The minimum steady-state control margin observed occurred during right sideward flight at the 20-knot envelope limit. At this condition, only 8.7 percent of the full pedal travel remained to the left. During acceleration to this condition, pedal movements came within 0.4 inch (5.3 percent) of the stop. Although test constraints allowed testing at only one weight and altitude, numerous tests (refs 33 and 34, app A) have shown that both left directional control and tail rotor power requirements increase as gross weight or density altitude increase. As power requirements increase in right sideward flight (with increases in gross weight or density altitude), even more left directional control will be required. Pilot workload increased as the aircraft approached translational lift. The greatest pilot workload was encountered at approximately 18 KTAS, left sideward flight, where large and frequent control movements were required about all axes and the left directional control stop was reached. At 20 KTAS, the pilot workload was reduced because the aircraft became more stable. It is anticipated that additional control problems will be encountered when the aircraft reaches translational lift with the associated large trim shifts. Directional control margins of less than 10 percent during stabilized sideward flight and contacting the directional control stops during transient sideward flight does not meet the intent of the requirements of MIL-H-8501A, in that adequate directional control is not provided. The inadequate directional control margin in sideward flight is a deficiency. Until the deficiency is corrected, the lack of adequate directional control margin warrants a reduction in the sideward flight envelope for future Army tests.

121. The 20-KTAS safety-of-flight release restriction prevented investigation of the translational lift regime at 20,500 pounds. Also, because of this restriction, compliance with the 35-knot sideward flight requirement and the 30-knot rearward requirement of MIL-H-8501A could not be investigated. Tail rotor power requirements could not be determined because of instrumentation problems. Further investigations should be conducted at maximum operational gross weight and altitude at various wheel heights to 35 KTAS in sideward flight and 30 KTAS in rearward flight.

WEAPONS FIRING

General

122. Aircraft S/N 66-8831 was used to conduct the weapons firing tests. The weapons fired included the XM51 (40mm), XM52 (30mm), and FFAR. The firing

was conducted at the conditions listed in tables 5, 6, and 7. The TOW missiles and the XM53 (7.62mm) weapon system were not fired because the test aircraft did not have these weapons systems installed.

Table 5. Firing Conditions, XM51 (40mm Nose Turret).

Indicated Airspeed (kt)	Gun Azimuth (deg)	Gun Elevation (deg)	Ammunition Expended (rd)
148	Zero	Zero	23
148	Zero	Full up	23
146	90 right	Zero	23
148	Traverse 30 right to 60 left	Zero	106
60	Zero	Zero	35
60	Zero	Full down	10
60	90 right	Full up	[1]31
60	90 right	Zero	[1]39
60	90 right	Full down	[1]49
Hover[2]	30 right	As required	7
Hover[2]	30 left	As required	30

[1]Fired in two bursts.
[2]Propeller thrust was used to obtain a nose-up attitude of approximately 10 degrees.

Table 6. Firing Conditions, XM52 (30mm Belly Turret).

Indicated Airspeed (kt)	Gun Azimuth (deg)	Gun Elevation (deg)	Ammunition Expended (rd)
170	Zero	Zero	14
180	Zero	Full up	15
175	Zero	Full down	14
185	90 left	Zero	13
180	90 left	Full up	15
175	90 left	Full down	15
150	Zero	Zero	19
150	90 right	Full up	14
150	180	Full up	14
150	Traverse zero to 90 right	As required	87
150	Traverse zero to 180 left	As required	122
140	90 right	Zero	26
140	180	Zero	20
60	90 left	Full up	14
60	90 left	Full down	15
60	90 left	Zero	6

Table 7. Firing Conditions, XM159 (2.75-inch FFAR).

Indicated Airspeed (kt)	Flight Path (deg)	Rockets Fired	Firing Mode
120	5 dive	6	Ripple
120	10 dive	6	Ripple
120	20 dive	6	Ripple
180	5 dive	6	Ripple
185	20 dive	14	Ripple
140	Level	38	Salvo

XM51 Weapon System

123. The firing tests of the nose-turret XM51 weapon system (40mm grenade launcher) produced the most noticeable aircraft response. In the forward azimuths and near zero elevations, neither aircraft response nor vibration was observed. As the weapon azimuth and/or depression were increased, the vibration levels at both crew positions increased. At the 90-degree azimuth positions, firing caused the nose of the aircraft to yaw away from the direction of fire if no pedal correction was made. This yawing was easily controlled with light pedal pressure at each airspeed evaluated. At 60 KIAS, precise control of this yawing was the most difficult, because of the reduced directional stability at this low airspeed, but was still considered acceptable for accurately firing this weapon. When firing at the full-down depression angles, a heavy vertical vibration at gun firing frequency was observed. This vibration was heavy enough to have a significant effect on target tracking at both crew positions and is a shortcoming.

124. During the firing, gun gas was detected in both crew stations on two firing runs (ref 35, app A). The first was at 60 KIAS (firing 20 rounds, 90 degrees right, and full-down depression). The second was during the 150-KIAS traverse firing test (approximately 105 rounds from 30 degrees right to 60 degrees left at zero elevation). In both cases, the environmental control unit (ECU) was ON, and the gas dissipated in 15 to 30 seconds. The presence of gun gases is a shortcoming. The noise level during the firing varied similarly to the changes in vibration level and was acceptable for all conditions evaluated. One abnormal shutdown and stoppage of the XM51 weapon system occurred during these tests as the result of an ammunition link failure.

XM52 Weapon System

125. The firing tests of the XM52 (30mm belly turret) weapon system caused much lower noise, vibration, and aircraft responses than the 40mm nose-turret weapon. Only in the zero elevation, 90-degree azimuth positions, was any significant aircraft response to weapon firing observed. These side firing positions caused a small roll and yaw response which was easily corrected and controlled. Stopping and starting firing in these side firing conditions produced no noticeable roll oscillations or control difficulties at any of the airspeeds evaluated. The most noticeable response related to the XM52 weapon occurred when the turret was abruptly moved in either azimuth or elevation in the direct control mode. These abrupt turret movements caused noticeable yaw and roll response from 5 to 6 degrees and from 3 to 4 degrees, respectively. These response characteristics were acceptable for the direct mode – a secondary or failure mode of operation of the XM52 turret. Aircraft response to turret movements made while operating in the primary (stabilized) mode of operation were less noticeable and were acceptable. Airframe vibration and noise observed during the XM52 weapon system firing tests were low. Strong gun-gas odor was detected during the 175-KIAS, zero-azimuth, full-down-depression firing run (ref 35, app A). The ECU was OFF at the time; however, the noticeable gas dissipated in 15 to 30 seconds. Abnormal shutdowns and weapon stoppages were experienced twice during these tests. The first resulted from a shear-pin failure in the aft drive motor. The second resulted from a separation of a conveyor chute link in the weapon feed system. During the test, 480 rounds of 30mm ammunition were fired. While traversing the weapon to the left, a single burst of 122 rounds was fired without interruption.

Folding-Fin Aircraft Rocket Firing

126. A full load of 38 rockets was fired at 140 KIAS in a single salvo without any noticeable effects either on the aircraft or the handling qualities. A second load was fired in dives of 5, 10, 15, and 20 degrees at approximately 120 and 180 KIAS. No aircraft response was observed while firing the rockets. The AH-56A characteristic which enables the pilot to stabilize on a selected airspeed at any dive angle was outstanding. Stable, trimmed 1.0g dives were easy to establish and maintain and should increase rocket accuracy. Difficulties in changing targets were observed during APE I and are discussed in paragraph 44. Controlling the free turbine and rotor speed during steep dive angles (greater than 20 degrees) and during the recovery phase may cause a significant increase in pilot workload since both tended to increase as the dive angle and amount of reverse Beta were increased. The effects of decreased propeller thrust on pilot workload should be further investigated to the limits of the reverse Beta control at all usable attack dive angles.

Hover Firing

127. The aircraft response to weapon firing while hovering IGE was evaluated. The XM51 weapon was fired while hovering approximately 20 feet above the ground in a 10-degree nose-high attitude. Thirty-seven rounds of 40mm ammunition were fired at 30-degree azimuths right and left of the nose with full-up elevation.

Although forward visibility from the pilot station was very restricted in this attitude, a stable hover was easily maintained during the firing. No unusual changes in aircraft attitude, vibration, or noise were observed during either of the firing bursts.

GROUND OPERATIONS

Preflight

128. The pilot's preflight inspection, as outlined in the preliminary operator's manual (ref 36, app A) was evaluated. The inspection appeared to be adequate and easily accomplished, except for the following areas:

a. Page 3-7, Paragraph 3-10. Add "Check tail wheel locking indicator and tail wheel in locked position."

b. Page 3-8, Paragraph 3-12. Add "Check fuel control shaft position to ensure that it is at the shut-off stop."

c. Page 3-8, Paragraph 3-12. Closing and securing the engine cowling requires two personnel with screwdrivers. It must be secured from both sides simultaneously and cannot be secured from the left side only after the inspection is completed. This condition is a shortcoming.

d. Page 3-8, Paragraph 3-13. A flashlight is required (even on a sunny day) to check the APU oil level and right-angle gearbox in the left gear well. The lack of adequate lighting is a shortcoming. A light to illuminate the oil level sight gages is desirable.

Ingress and Egress

129. Ingress and egress in both cockpits were satisfactory except for two shortcomings:

a. The lack of a handhold on the aft right sponson above the boarding ladder to aid combat-equipped aviators to board the aircraft.

b. The lack of a handrail on the top of the fuselage between the engine inlet and the aft canopy.

Start and Run-Up

130. The auxiliary power unit (APU) starting procedure is automatic. However, the reliability of the control box was low and is a shortcoming. The APU failed to start 12 times out of 30 attempts, and the control box was changed six times during the APE I.1 test program. (ref 37, app A). Numerous failures occurred during APE I.3, and the control box was changed three times.

131. The engine start is automatic after the pilot activates the starter button and moves the engine condition grip from the OFF to the IDLE position at a 20-percent gas producer speed.

132. The run-up and systems checks are satisfactory.

Taxi

133. Ground taxi with propeller thrust instead of the usual cyclic stick movement was outstanding: the aircraft handles very well on the ground. Maximum reverse thrust propeller angle of the APE I test aircraft was -5 degrees instead of the design -17.2 degrees because of instrumentation. For this reason, brakes were required to assist in stopping. The brakes were effective and not overly sensitive. These brakes have been observed to fade after heating up from hard braking. This fading is a problem area and is a shortcoming. Directional control during taxi was excellent. Vibration levels in the front seat were low, and in the rear seat the vibration levels were moderate. The tail wheel must be locked by a lever in the cockpit for takeoff and landing. However, the tail wheel must be unlocked for all taxi turns or it will be subjected to side loads strong enough to skid the tire. On several occasions, the tail wheel remained locked even though the lever had been activated to unlock it. The pilot cannot detect this condition until excessive tire skidding has occurred. The lack of a reliable tail-wheel unlocking mechanism is a shortcoming.

COCKPIT EVALUATION

134. The cockpit of the APE I test aircraft was nonstandard with special test instrumentation installed; however, the following items, applicable to all AH-56A aircraft, were noted:

a. The toggle switches for the hydraulic systems, the ALTERNATE N_f, ignition, boost pump, anti-icer, engine air, and environmental control system are miniature switches that are easily bent, a shortcoming. Two switches, IGNITION and ALTERNATE N_f, were found bent during pilot training and this test. These switches should be the stronger type toggle switches used for the other cockpit switches.

b. The propeller and engine condition grips are the same configuration, a shortcoming. These grips are located on the collective lever with the propeller grip forward and the engine condition grip immediately behind it. The engine condition grip can easily be mistaken for the propeller grip and inadvertently reduced to IDLE in flight. There should either be a positive lock on the engine condition grip in the RUN position, or it should be relocated to avoid confusion.

c. The emergency fuel and engine shutoff system is electrical, which presents a safety hazard if all electrical power is lost and emergency fuel and engine shutoff is required. The lack of a mechanical fuel and engine shutoff in the cockpit is a shortcoming.

d. The ALTERNATE N_f increase/decrease control is overly sensitive. Operation of the ALTERNATE N_f in the decrease direction can quickly decrease rotor speed below operating limits and shut down the engine. The excessive sensitivity of the ALTERNATE N_f control is a shortcoming.

e. The rotor brake is an excellent feature and very effective in stopping the rotor.

f. An accurate accelerometer must be provided in all AH-56A aircraft to assist the pilot to remain within the prescribed load factor envelope. Without an accelerometer, adequate identifiable pilot cues are not available.

g. The fault locator aural warning system (FLAWS) and the voice warning system are excellent concepts and should be included on all future AH-56A aircraft to warn the pilot of malfunctions while he is heavily engaged in flight, navigation, and fire control system tasks. This system, however, must be accurate and reliable. Nuisance activations occurred five times in 18 flights during APE I.1. The reliability of this system was improved during APE I.3.

h. The lack of a placard or gage markings showing collective blade angle limits is a shortcoming.

i. The circuit breaker panels are above and behind the pilot's head. They are difficult to see (requiring 180-degree head rotation) and most cannot be reached by a pilot with average length arms while strapped in his seat. The inability to reach most of the circuit breakers in flight is a shortcoming.

j. Either the pilot or the copilot/gunner has the capability of extinguishing both "MASTER CAUTION" warning lights (ref 38, app A). When the copilot/gunner depresses his "MASTER CAUTION" light, the pilot's warning is also extinguished. This is undesirable because the pilot has only a limited number of items on his caution panel, and the pilot may not be alerted to the warning condition. Depressing the "MASTER CAUTION" warning light should only extinguish the one light being depressed. Correction of this shortcoming is desirable.

k. The government-furnished radios were not equipped with the capability to preset frequencies, and an excessive amount of pilot attention and time was required to manually set frequencies. The ability to quickly shift UHF, VHF, and FM frequencies is essential to accomplishment of attack helicopter tasks involving coordination of air strikes, artillery fires and several ground maneuver elements. The lack of capability to rapidly shift to preselected frequencies is a shortcoming.

135. The ECU is necessary for equipment and personnel cooling. During this test, use of the ECU was required for cockpit cooling, even when the outside air temperature was below 20°C/68°F because no other source of ventilation was available. At higher temperatures, the ECU was inadequate to provide crew comfort, a shortcoming. The ECU was generally effective in removing smoke and fumes that were present in the cockpit on several occasions during the test. The ECU

operation increased the cockpit noise level which occasionally interfered with ICS communications and radio transmissions, a shortcoming.

MISCELLANEOUS

General

136. This section of the report discusses the airspeed calibration, vibration characteristics, structural loads, control system characteristics, and maintenance items.

Airspeed Calibration

137. Airspeed calibration tests were conducted during APE I.1 to determine the position error of the standard ship's system and the test (boom) airspeed system in level flight. A pacer aircraft (F-51D) was used for the calibration. The calibration was conducted in the clean configuration. The data are presented in figures 88 and 89, appendix F.

Vibration Characteristics

138. During APE I.1, vibration data were recorded in the vertical and lateral axes in the front cockpit (FS 124) and the aft cockpit (FS 170). Vertical vibration data were also recorded near the aircraft cg (FS 310). Vibrations at frequencies corresponding to the main rotor harmonics (1/rev, 2/rev, 4/rev, and 8/rev) were evaluated. Quantitative vibration data during maneuvers were not obtained due to lack of adequate instrumentation. Additionally, the 1/rev and 2/rev vibration data are questionable because the response of the vibration sensors was not linear at these frequencies. These data are presented in figures 90 through 94, appendix F.

139. Vibrations were measured from 38 to 189 KCAS at the stations listed in paragraph 138. The aft seat lateral vibrations were low throughout the speed envelope. At both crew locations, the 4/rev vertical vibration levels were generally below the specification limits at airspeeds below 184 KCAS. The 8/rev vibration levels at these locations were generally double the maximum specification limits above 110 KCAS rising sharply to triple the maximum specification limit at 189 KCAS. The 4/rev vertical vibrations in the front cockpit were low compared to the lateral vibrations which ranged from moderate to objectionable. Just the opposite was true in the aft cockpit where the lateral vibrations were low and the vertical vibrations ranged from low to objectionable. As airspeed increased for a given collective setting, the vibration levels generally increased. When the collective was lowered, the vibration level decreased. Recorded 4/rev and/or 8/rev vibration levels throughout the flight envelope generally exceeded the limits of MIL-H-8501A (hence CP0001A) at all locations, except the aft seat lateral. In some cases, these vibration levels were two to three times the maximum military specification limits. Vibration levels during hover in calm winds were relatively low. Wind direction and speed had a marked effect on vibration levels during hover and hover takeoffs.

On one occasion during hover with a right rear quartering tail wind of 12 to 18 knots, vibration levels from the main rotor, propeller, and tail rotor were so high that the pilot was concerned that something was wrong with the aircraft and aborted the takeoff. The levels subsided when the aircraft was turned into the wind. Accelerations from and decelerations to a hover resulted in objectionally high vibration levels which on some occasions made the instruments unreadable and caused a numbing sensation in the pilot's body. This degraded the pilot's ability to perform necessary tasks.

140. Prior to APE I.3, the right wing and horizontal stabilizer incidence were changed to reduce the cyclic blade angle requirements at high speed in an attempt to reduce the vibration. Only limited vibration data were obtained because of instrumentation problems. Data obtained during the static longitudinal and lateral-directional stability tests are presented in figures 25 through 32, appendix H. Qualitatively, there appeared to be a reduction in the 4/rev and 8/rev vibration amplitudes at speeds above 160 knots; however, the requirements of CP0001A were still exceeded. There was a definite and perceptible increase in 4/rev vibration with sideslip, particularly to the right. Qualitatively, vibration levels in aircraft S/N 66-8831 were lower than in aircraft S/N 66-8834. The excessive 4/rev and 8/rev vibration level in the test aircraft S/N 66-8834 exceeded the specification limits, degraded the pilot's ability to perform necessary tasks, caused undue pilot fatigue, and is a deficiency.

Structural Loads

141. Structural loads data were recorded continuously during the test program. The results of these tests will be presented in a supplement by AVSCOM.

Control System Characteristics

142. The front cockpit control breakout forces, force gradients, and ranges of movement were determined during ground tests with the rotors stationary. Ground power units were used to provide electrical and hydraulic power for the tests. Only the number-two hydraulic system functioned during ground power operations. Control forces were measured from the center of the cyclic grip, the base of the directional pedals, and at the center of the propeller twist grip control for the collective. Control positions were measured using the onboard instrumentation. Breakout forces (including friction) were determined by recording the forces required to obtain initial movement of each control. The force gradients were determined from plots of the control position versus applied increment of force. Measurements were made in both directions from the trim null position for each control. The tests were conducted during APE I.1 and repeated during APE I.3B. The data from the tests are presented in figures 95 through 99, appendix F, and figures 41 through 46, appendix I. The results are summarized in table 8.

Table 8. Control Forces and Force Gradients.

Rotors: stationary

Parameter	"Q" Sensor Airspeed (kt)	Minimum/Maximum Allowed by MIL-H-8501A	Test Results APE I.1	Test Results APE I.3B
Longitudinal breakout[1]	Zero	N/A	16.0 lb	N/A
Longitudinal breakout[2]	Zero	0.5/1.5 lb	2.0 lb (aft) 1.0 lb (fwd)	2.0 lb (aft) 2.5 lb (fwd)
	100	N/A	N/A	1.0 lb (aft) 2.5 lb (fwd)
	150			2.5 lb (aft) 1.5 lb (fwd)
	189		0.0 lb (aft) 2.0 lb (fwd)	N/A
	200		N/A	1.0 lb (aft) 2.5 lb (fwd)
Longitudinal gradient[2]	Zero	0.5/2.0 lb/in.	4.0 lb/in.	4.0 lb/in. (aft) 3.0 lb/in. (fwd)
	100	N/A	N/A	5.5 lb/in. (aft) 3.5 lb/in. (fwd)
	150			7.0 lb/in. (aft) 4.5 lb/in. (fwd)
	189		6.5 lb/in.	N/A
	200		N/A	5.0 lb/in. (aft) 3.5 lb/in. (fwd)
Lateral breakout[1]	N/A	N/A	10.0 lb (right) 9.0 lb (left)	N/A
Lateral breakout		0.5/1.5 lb	2.0 lb	1.0 lb
Lateral gradient[1]		N/A	2.5 lb/in.	N/A
Lateral gradient[2]		0.5/2.0 lb/in.	2.0 lb/in.	1.5 lb/in.
Pedal breakout		3.0/7.0 lb	8.0 lb (right) 7.0 lb (left)	7.0 lb (right) 5.0 lb (left)
Pedal gradient		N/A	11.0 lb/in. (right) 11.5 lb/in. (left)	10.0 lb/in.
Collective breakout[3]			15.0 lb	14.0 lb (up) 12.0 lb (down)
Collective breakout[4]			23.0 lb	19.0 lb (up) 20.0 lb (down)
Collective breakout[5]			7.0 lb	N/A
Collective breakout[6]			17.0 lb	
Collective downstop breakout[5]			15.0 lb to 22.0 lb	

[1]Stick centering: ON.
[2]Stick centering: OFF.
[3]Electric friction: ON. Mechanical friction: minimum.
[4]Electric friction: ON. Mechanical friction: maximum.
[5]Electric friction: OFF. Mechanical friction: minimum.
[6]Electric friction: OFF. Mechanical friction: maximum.

143. The longitudinal trim actuator travel during APE I.1 was equivalent to 7.2 inches of control travel at a constant rate of 0.90 inch per second (in./sec). The lateral trim actuator travel was equivalent to 3.55 inches of stick movement with a rate of 0.5 in./sec which was constant throughout its travel. The directional trim actuator travel was not checked during APE I.1. During APE I.3B, the full travels for longitudinal, lateral, and directional trim actuators were equivalent to 7.2, 4.4, and 6.4 inches of control travel, respectively. Trim rates were 0.9 in./sec, longitudinal, 0.71 in./sec, lateral, and 0.41 in./sec, directional.

144. During flight operations, two shortcomings of the control system were encountered: the difficulty in trimming lateral forces to zero, and the inability of the directional beeper trim to keep up with the large directional trim changes that occur during transition from hover to forward flight and from forward flight back to hover (paras 28, 29, and 98). These shortcomings were apparent during all the tests in both aircraft.

145. Another shortcoming was found during ground tests. The lateral trim actuator position affects the lateral cyclic control travel. During APE I.1, when full lateral trim was applied in one direction, the lateral control travel in the opposite direction was reduced by about 1 inch. This restriction was considered a deficiency because flight conditions existed where loss of control could result from the restricted control travel (fig. 12, app F). The problem was corrected and was not encountered during APE I.2 but was again apparent during APE I.3B because of the lateral control modifications. The amount of restriction during APE I.3B was less (0.6 inch right, and 0.36 inch left), and the problem is now a shortcoming.

Maintenance

146. The following maintainability and reliability shortcomings were noted during the conduct of these tests:

a. The main landing gear oleo struts were quite uneven before and/or after almost every flight during the test program (ref 25, app A). The contractor was not able to satisfactorily correct this problem. It is desired that the oleo struts compress evenly when the aircraft is on the ground.

b. The squib which activates the pneumatic emergency landing gear extension system can be inadvertently fired by cycling the switch in the flight control bay during maintenance. This condition may not be detected prior to flight. This situation was noted during APE I.1 and APE I.2. Prior to APE I.3, a safety wire was installed which decreased the possibility of inadvertent squib firing. Check of the accumulator pressure should be included in the daily inspection. An emergency landing gear extension system which will operate at airspeeds up to 130 KCAS should be included in the production aircraft.

c. The main transmission oil tank dip stick is unreliable because it does not correctly indicate the total oil in the system (ref 39, app A). Because of this, the transmission was often overfilled, possibly resulting in cockpit fumes which occurred on numerous occasions during this test. An accurate and reliable main transmission oil quantity sight gage should be provided.

d. Tracking methods used on the main rotor and balance methods used on the tail rotor during this test program would be unacceptable as field procedures. The procedures used were dependent upon the large amount of test instrumentation on the aircraft. A field procedure for tracking of the main rotor and balance of the tail rotor should be provided.

e. Daily inspection of the gyro hub and main rotor requires using the cooling air vent to climb up to this location, and requires standing on the unreinforced fuselage skin above the electronics compartment. This has caused damage to the screen in the cooling air vent and undue stress to the fuselage skin. Steps and a strengthened work area should be provided to allow inspection and maintenance on the main rotor and gyro hub.

f. In order to inspect or perform maintenance inside the engine inlet filter cowling, the cowling must be removed entirely or supported by rods which are not an integral part of the aircraft. An integral support should be provided.

g. To check or service the main landing gear emergency air bottle and the rotor brake accumulator, a stress panel secured by 17 structural fasteners must be removed. This makes preflight and postflight servicing and inspection difficult and increases turnaround time.

h. The tail rotor assembly leaked oil excessively, especially around the garlock seals on the blade grips.

i. Excessive wear on the right side of both tires resulted from scuffing continuously during taxiing. The wear rate required that the tires be switched after approximately 20 landings and replaced after approximately 40 landings.

j. Replacement of the propeller gearbox was observed to require several days, special tooling, and was difficult. The difficulty appears to be in mating the gearbox to its mounting surface. Proper manufacturing tolerances and reference points for proper measurement and alignment should be provided for the production configuration.

k. The bonding of stainless-steel, silver-plated shims to the inside upper grip of the titanium moveable hub was inadequate. The shims (one to three of them) came loose every time the outboard tension/torsion pack bolts were inspected (every 15 hours).

l. Present contractor procedure is to change air pressure in the tires with each change in aircraft gross weight. This procedure is an unacceptable field requirement.

m. Indicated fuel quantity was as much as 400 pounds different from actual quantity, and indicator calibrations changed with time. The fuel quantity gage was unacceptably inaccurate due to hysteresis, coarse graduations, and nonrepeatability.

n. The transmission oil pressure gage was inaccurate.

o. During these tests, numerous fatigue cracks occurred throughout the tail boom structure indicating structural inadequacy of the tail boom.

CONCLUSIONS

GENERAL

147. The following conclusions were reached upon completion of the AH-56A Army Preliminary Evaluation I and the USAASTA portion of the AH-56A Research and Development Acceptance Test I:

a. The handling qualities of the AH-56A were not significantly affected by firing of any armament subsystems tested.

b. Lateral stick migration with airspeed was not objectionable (para 23).

c. Maintenance of trim airspeeds was enhanced by the pusher propeller (para 37).

d. Stick-free maneuvering stability was improved by the increased bobweight design effectiveness but was still nonlinear (para 53).

e. Lift/roll coupling has been reduced by the lift/roll decoupler but is still objectionable (paras 55 and 81).

f. Two deficiencies which exist within the presently approved test flight envelope of the AH-56A warrant a reduction of envelope size for future Army tests (paras 61 and 121).

g. With the pitch desensitizer system installed, the excessive longitudinal control sensitivity observed during APE I.1 is no longer objectionable (para 67).

h. The lateral control response has been reduced from the excellent levels available during APE I.1 and APE I.2 (para 71).

i. Pitch/roll coupling has been virtually eliminated because of the pitch/roll decoupler (para 83).

j. Roll damping with the roll compensator OFF has decreased since APE I.1. Hover takeoffs and landings should not be attempted with the roll compensator inoperative (para 92).

k. With the roll compensator ON, the roll damping is satisfactory (para 92).

l. The rotor dynamic instabilities previously encountered in the contractor development program were not apparent at the test flight conditions (paras 94 and 95).

m. The ability to stabilize in various pitch attitudes in a hover is a unique and desirable feature (para 96).

n. Safe operations to and from unimproved areas will require training emphasis in the use of propeller pitch control (para 99).

o. Tendency towards pilot-coupled roll oscillation has been significantly reduced, but the tendency to overcontrol is still present throughout the flight envelope, particularly during hover takeoffs and landings (HQRS 4) (para 111).

p. The roll damping appeared to deteriorate with increasing gross weight (para 111).

q. Power management in the AH-56A is more complicated than in conventional helicopters (para 112).

r. The ability to decelerate rapidly using reverse propeller thrust is an excellent feature (para 118).

s. The ability to maintain stabilized airspeed independent of dive angle is an excellent characteristic (paras 118 and 126).

t. Ground taxi with propeller thrust is excellent (para 133).

u. The rotor brake is an excellent feature (para 134e).

v. Reliability of the FLAWS and voice warning system has been improved (para 134g).

w. The FLAWS and voice warning system are excellent concepts (para 134g).

x. Low reliability of several components as well as overall maintenance requirements were generally unsuitable for field use (para 146).

y. Throughout the latter phases of the test program, improvements in many areas related to early deficiencies and shortcomings were noted; however, five deficiencies and 54 shortcomings remain uncorrected at the completion of the testing.

DEFICIENCIES AFFECTING MISSION ACCOMPLISHMENT

148. Correction of the following deficiencies should be a prerequisite for an airworthiness release for operational Army aviators:

a. The requirement for excessive pilot workload due to unacceptable static lateral-directional stability characteristics below 100 KIAS seriously impairs the capability to operate at minimum altitudes unaffected by conditions of darkness or adverse weather (para 47).

b. Uncommanded aircraft motion and loss of control during maneuvering flight (HQRS 10) (para 61).

c. The rapid rate of rotor speed decay following simulated engine failures which allows the rotor to drop below the present transient limit (para 116).

d. Inadequate directional control margin in sideward flight (para 120).

e. Excessive 4/rev and 8/rev vibration levels (para 140).

SHORTCOMINGS AFFECTING MISSION ACCOMPLISHMENT

149. Correction of the following shortcomings is desirable:

a. Uncomfortable rigging of the right pedal which contributed to difficulty in maintaining zero sideslip (para 24).

b. Considerable pilot attention required for longitudinal trim at high airspeeds (para 27).

c. Excessive pilot attention required for lateral trim (paras 28 and 144).

d. Excessive attention required for operation of the directional trim during transition (para 29).

e. Unsatisfactory collective control friction characteristics (para 30).

f. Inability of the propeller control system friction to maintain desired Beta settings (para 31).

g. Weak static longitudinal stability at 150 KCAS and above (para 38).

h. Lack of adequate directional stability which made it very difficult to establish and maintain a desired heading and sideslip (para 40).

i. Weak side-force characteristics (para 42).

j. Excessive inherent sideslip at high airspeeds (para 43).

k. Excessive longitudinal trim shift with sideslip (pitch due to sideslip) (paras 44, 57, and 78).

l. Objectionable longitudinal control force during sideslips (para 45).

m. Poor stick-fixed maneuvering stability (para 51).

n. Objectionable lift/roll coupling (paras 56 and 81).

o. Disharmony between longitudinal and lateral control displacements and forces (para 56).

p. Lack of satisfactory warning of uncommanded pitch-up and blade moment stall (para 59).

q. Inability to perform operational maneuvers at high speed (para 60).

r. Excessively long pitch time constant (para 64).

s. Roll oscillations with changing load factor (para 81).

t. Failure of the main landing gear struts to compress evenly during landing (para 103).

u. Tendency to overcontrol during hover takeoffs and landings (HQRS 4) (para 111).

v. Excessive pilot attention required for power management at high speeds (para 113).

w. Heavy vertical vibrations when firing the XM51 weapons system (para 123).

x. Presence of objectionable gun gases when firing the XM51 and XM52 weapons systems (paras 124 and 125).

y. Lack of an acceptable procedure for closing and securing the engine cowling (para 128c).

z. Lack of adequate lighting for APU compartment oil-level inspection (para 128d).

aa. Lack of a handhold on the aft right sponson above the boarding ladder (para 129a).

ab. Lack of a handrail on top of the fuselage between the engine inlet and aft canopy (para 129b).

ac. Low reliability of the APU control box (para 130).

ad. Brake fading after hard application (para 133).

ae. Lack of a reliable tail wheel unlocking mechanism (para 133).

af. Easily bent miniature switches used for primary functions in the cockpit (hydraulic systems, ALTERNATE N_f, ignition, boost pump, anti-icer, engine air, and ECU) (para 134a).

ag. Identical configuration and proximity of engine and propeller twist grips (para 134b).

ah. Lack of a mechanical fuel and engine shutoff that can be operated from the cockpit (para 134c).

ai. High sensitivity of ALTERNATE N_f increase/decrease switch (para 134d).

aj. Lack of a collective schedule placard or gage markings (para 134h).

ak. Inability to reach most circuit breakers in flight (para 134i).

al. Ability of either pilot or copilot/gunner to extinguish both "MASTER CAUTION" lights (para 134j).

am. Lack of capability to rapidly change radio frequency (para 134k).

an. Inadequate cooling capacity of the ECU (para 135).

ao. Interference with ICS communications and radio transmissions by ECU operation (para 135).

ap. Restriction of lateral control travel with full lateral trim (para 145).

aq. Uneven main landing gear oleo struts before and/or after flight (para 146a).

ar. Lack of an accurate and reliable main transmission oil quantity sight gage (para 146c).

as. Lack of an acceptable field procedure for tracking of main rotor and balance of tail rotor (para 146d).

at. Lack of step and reinforced work area for main rotor and gyro hub maintenance and inspection (para 145e).

au. Lack of integral support for engine inlet filter cowling when opened for inspection or maintenance (para 146f).

av. Requirement to remove a stress panel secured by 17 stress fasteners to inspect and service the main landing gear emergency air bottle and rotor brake accumulator (para 146g).

aw. Excessive oil leakage from the tail rotor assembly (para 146h).

ax. Excessive wear of tires during ground operations (para 146i).

ay. Difficulty in accomplishing propeller gearbox changes due to improper mating surfaces (para 146j).

az. Inadequate bonding of the stainless-steel, silver-plated shims to the inside upper grip of the titanium moveable hub (para 146k).

ba. Unacceptable requirement to change the pressure with changes in gross weight (para 146l).

bb. Inaccurate fuel quantity gage (para 146m).

SPECIFICATION NONCONFORMANCE

150. Program constraints did not permit the determination of conformance with all applicable paragraphs of military specifications MIL-H-8501A and MIL-F-8785(ASG).

151. Within the scope of these tests, the AH-56A failed to meet the following requirements of military specification MIL-H-8501A:

a. Paragraph 3.2.1 - Rearward flight at 30 knots was outside the flight envelope (para 121).

b. Paragraphs 3.2.10 and 3.6.3 - The longitudinal control position and control force stability with respect to airspeed were not stable at all airspeeds tested (para 37).

c. Paragraph 3.3.2 - Sideward flight at 35 knots was outside the flight envelope (para 121).

d. Paragraph 3.3.9 - The variations of pedal displacement and lateral control displacement with steady sideslip angle were not stable at all the speeds specified. Additionally, the minimum sideslip angles specified in this paragraph were outside the envelope (paras 40 and 41).

e. Paragraph 3.3.15 - A tendency to overcontrol unintentionally in roll was present (para 111).

f. Paragraph 3.5.5 - The intent of this paragraph was not complied with, in that a 2-second delay in the movement of power controls following simulated engine failures would result in excessive rotor speed decay (para 116).

g. Paragraph 3.6.1 - Undue pilot effort would be required to perform instrument flight (para 89).

h. Paragraph 3.7.1 - The aircraft was not free of objectionable shake vibration or roughness, and maximum vibration amplitude accelerations specified were exceeded at many conditions during these tests (para 139).

152. Within the scope of these tests, the AH-56A failed to meet the following requirements of military specification MIL-F-8785(ASG):

a. Paragraph 3.3.4 – Neutral stick-fixed maneuvering stability, as evidenced by the zero slope of longitudinal control position versus load factor (paras 49 and 51).

b. Paragraph 3.3.9 – The slope of the curve of longitudinal control force versus load factor failed to meet the linearity requirements of this paragraph (para 52).

c. Paragraph 3.3.20 – The longitudinal trim change with sideslip encountered during these tests did not meet the requirements of this paragraph, in that the trim change was not in the same direction with right and left sideslips, and the push force required in right sideslips exceeded the 3-pound limit specified (para 45).

RECOMMENDATIONS

153. The deficiencies should be corrected prior to release to operational Army aviators.

154. Correction of the shortcomings is desired.

155. Additional testing should be conducted by the contractor to evaluate the blade moment stall characterisitcs up to the structural or control limits (para 58).

156. High-speed maneuvering capability should be increased (para 60).

157. Further testing should be conducted to optimize roll response for the attack mission (para 71).

158. The effect of component deterioration on lateral roll oscillations should be determined (para 91).

159. Hover takeoffs and landings should not be attempted with the roll compensator inoperative (para 96).

160. Pilot training must emphasize the unique characteristics of varying propeller thrust and its effect on safe operation to and from unimproved areas and uneven terrain (para 99).

161. Pilot training should emphasize the increase in power required with decrease in airspeed at fixed collective and propeller blade angles (para 113).

162. A complete investigation of the autorotational entry, descent, and landing characteristics should be conducted by the Army (para 116).

163. Flight testing should be conducted at maximum operational gross weight to 35 knots in sideward flight and 30 knots in rearward flight (para 121).

164. The effects of decreased propeller thrust on pilot workload during dives should be further investigated (para 126).

165. An accurate accelerometer should be installed in all production AH-56 aircraft (para 134f).

166. An emergency landing gear system extension that will operate at airspeeds up to 130 KCAS should be provided in all production AH-56 aircraft (para 146b).

167. A design review should be conducted to determine the structural adequacy of the tail boom (para 146o).

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36. Preliminary Operational/Maintenance Manual, POMM 15-1520-222-10, *Operator's Manual, Helicopter, Attack, AH-56A (Lockheed),* July 1971.

37. Equipment Performance Report, USAASTA, SAVTE-T, EPR 70-02-01, Project No. 70-02, "APU Control Box," March 1971.

38. Equipment Performance Report, USAASTA, SAVTE-T, EPR 70-02-10, Project No. 70-02, "Master Caution System," 16 December 1971.

39. Report, Lockheed-California Company, *Monthly Test Progress Report Number 58,* May 1971.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The AH-56A Cheyenne is a two-place compound attack helicopter. Power is provided by a single General Electric T64-GE-16 (S4C) engine rated at 3925 shp maximum at sea level on a standard day. The main rotor, pusher propeller, and tail rotor share the engine power. Lift is provided by a combination of the main rotor and the wings. The wings provide an increasing proportion of lift with increasing airspeed. Attitude control is accomplished by the main rotor and the tail rotor, as no control surfaces are built into the wings or empennage.

2. Distinctive features of the AH-56A include the rigid-type four-bladed main rotor, a tail-mounted pusher propeller, low wings, conventional retractable landing gear, and a vertical stabilizer mounted below the fuselage. Sponsons are mounted along each side of the fuselage and house fuel tanks, the retracted main landing gear, an auxiliary power unit, an environmental control unit, and the fueling station. The tail wheel retracts into the vertical stabilizer.

3. The cockpit provides tandem seating for the pilot and the copilot/gunner. Standard configuration is for the pilot to fly the aircraft from the rear seat and for the copilot/gunner to operate the swiveling gunner station (SGS) in the front cockpit. This configuration was provided for the RDAT I test aircraft (S/N 66-8831). The APE I test aircraft (S/N 66-8834) differs from this configuration, in that the pilot station is in the front cockpit due to the installation of a downward ejection seat required for the contractor's developmental testing.

4. Provisions are made for both internal and external armament in the design of the AH-56A. Internal armament consists of the XM52 area fire system in the belly turret and either the XM51 or XM53 suppressive fire system in the nose turret. The XM52 system includes the XM140 (30mm automatic gun), the XM51 system includes the XM129 (40mm grenade launcher), and the XM53 system includes the XM134 (7.62mm minigun). Each of these weapons can be fired by either the pilot or the copilot/gunner. Six external pylons are provided for carrying armed stores and/or external fuel tanks. The two fuselage pylons are equipped to carry fuel tanks. The four wing pylons may be used to carry a variety of combinations of stores, including TOW missiles, 2.75-inch folding-fin aircraft rockets (FFAR), or external fuel tanks. The pilot may utilize the helmet sight system firing the flexible weapons, or he may use a direct sight for firing the forward firing weapons. The copilot/gunner may use the SGS periscopic sighting system, which includes a laser rangefinding system, for precise sighting and target tracking of the flexible weapons. In addition, an optical display sight is provided for target acquisition and coarse target tracking. The computer central complex (CCC) provides ballistics corrections and prediction calculations for the weapons systems. The APE I test aircraft was not configured with the weapons systems.

CONTROL SYSTEM

5. Conventional helicopter controls are provided, utilizing a cyclic stick for pitch and roll control, a collective lever for lift control, and pedals for directional control. The reversible pitch pusher propeller is controlled by means of a twist grip mounted on the collective lever. The cyclic, collective, and tail rotor control systems utilize dual tandem-servo actuators to amplify and transmit pilot or gunner control inputs to the control surfaces. Cyclic control inputs are transmitted by the servos to a positive spring and to the swashplate. The positive spring converts the control displacement to a force that is transmitted from the swashplate to the control gyro. The force produces a moment which causes the gyro to precess, providing cyclic blade angle changes. Swashplate feedback is provided to the roll servo actuator to reduce cross coupling due to gyro pitch precession. This swashplate feedback feature has been previously referred to as a "negative spring." Collective control movements are transmitted to the swashplate through a servo which moves the swashplate up and down, causing the control gyro to move vertically on the rotor shaft axis, producing blade angle changes simultaneously to all four blades. Prior to APE I.3B, detuning weights were installed to the collective control lever in the rear cockpit of S/N 66-8834. A force feel system is incorporated in the pitch, roll, and yaw control systems to provide simulated feel as the control is displaced from the selected trim position. Trim systems are provided to relieve the feel forces when the control is held out of neutral.

6. The pitch control system includes four augmentation devices intended to improve AH-56A handling qualities. These devices are identified as the velocity gradient, maneuver gradient (bobweight), pitch desensitizer, and pitch/roll decoupler systems. The velocity gradient and maneuver gradient systems operate within the longitudinal feel system and provide increasing stick forces with increasing airspeed and load factor, respectively. The maneuver gradient was changed from 6 lb/g to 9 lb/g prior to APE I.3. The pitch desensitizer system reduces the longitudinal control response and sensitivity at high speed. This system senses airspeed and longitudinal control displacement from trim to determine the size of control input required. The control input is made through a modulation piston in the pitch servo and is not felt by the pilot. An airspeed-scheduled gain signal to the desensitizer system is zero for airspeeds at or below 100 knots and varies linearly to full gain at 170 knots. At full gain, the system doubles the pilot longitudinal control displacement required to obtain a given aircraft response. Maximum authority of the system is equivalent to ±0.757 inch of longitudinal stick displacement. The fourth augmentation device was designed to reduce pitch-due-to-roll cross-coupling. This system applies longitudinal control inputs through the desensitizer modulation piston to oppose the pitching moment caused by aircraft roll rates. The system senses airspeed and roll rate to tailor the size of control input applied. The gain signal from the airspeed sensor is zero at speeds up to 110 knots and varies linearly to full gain at 200 knots. The gain signal from the roll rate gyro reaches a maximum at 30 deg/sec. Therefore, at 200 knots and 30 deg/sec of right roll rate, the system will apply the full authority of the longitudinal piston (equivalent to 0.757 inch of aft longitudinal stick).

7. The pitch desensitizer and pitch/roll decoupler were not installed for APE I.1 and APE I.2. Both systems were installed and activated prior to APE I.3 and RDAT I.

8. For APE I.1 and APE I.2, the lateral control system featured a roll desensitizer which was designed to reduce roll control power, hence aircraft sensitivity, for control movements within 3/4 inch of the trim point. This system was removed prior to APE I.3.

9. The lateral control system incorporates a stability augmentation system (roll SAS) known as the roll compensator which was designed to increase the damping of roll oscillations at a 1-hertz frequency. The roll SAS applies control inputs through a modulation piston in the roll servo which opposes the rolling motion of the aircraft. The gain varies as a function of airspeed and of the frequency and magnitude of aircraft roll oscillations. The phasing between aircraft roll oscillations and roll SAS control inputs varies as a function of roll oscillation frequency. Maximum authority of the system is equivalent to approximately ±0.329 inch of lateral control displacement. Prior to APE I.3B, two notch filters were added to suppress 16-hertz and 32-hertz vibratory inputs to the roll SAS.

10. Another feature of the AH-56A lateral control system is a lift/roll decoupler which is intended to eliminate lateral control input changes in maneuvering load factor. This system was inoperative prior to APE I.3B testing.

11. A number of other control system changes were made during APE I.3A and before APE I.3B. These included:

a. Reduced control system free play.

b. Reduced lateral control power (control moment per inch of control displacement).

c. Increased lateral control travel (to provide the same total control moment).

d. Increased swashplate feedback.

e. Increased roll modulation piston stroke (to provide adequate lift/roll decoupler authority).

f. Reduced lateral control breakout forces.

g. Reduced lateral force gradient.

12. Principal control system characteristics are tabulated below:

<u>Cyclic Control System</u>

Gyro designation	1019896
Gyro polar moment of inertia	45 slug-ft^2
Gyro diameter	9.7 ft
Gyro arm diameter (gyro station 9.510 to gyro station 10.510)	2.55 in.
Gyro arm taper ratio, gyro station 10.51 to tip	0.0036 in./in.
Gyro arm incidence	Zero deg
Gyro cant angle	33 deg
Gyro maximum tilt angle	±15 deg
Design stick throw:	
Longitudinal	11.0 in.
Lateral	6.0 in. (APE I.1, APE I.2, APE I.3A, RDAT I) and 7.5 in. (APE I.3B)
Control input rotation	36.0 deg
Gyro moment per inch of stick:	
Longitudinal	278 ft-lb/in.
Lateral	450 (60%) ft-lb/in. (APE I.1, APE I.2, APE I.3B, RDAT I) and 337 (45%) ft-lb/in. (APE I.3B)
Net spring restraint per radian of gyro travel:	
Longitudinal	4100 ft-lb/rad
Lateral	4100 ft-lb/rad

Gyro damping per damper (2 pitch, 2 roll)	44 in.-lb/rad/sec
Total feather bearing friction at gyro	28 ft-lb (approx)
Moment at gyro due to total nonrotating system friction	30 ft/lb (approx)
Servo rate:	
Longitudinal	5.62 in./sec
Lateral	5.62 in./sec
Trim authority:	
Longitudinal	70 percent
Lateral	70 percent
Stick damping (at grip):	
Longitudinal	0.167 lb/in./sec (hover) and 0.28 lb/in./sec (225 kt)
Lateral	0.115 lb/in./sec (hover) and 0.115 lb/in./sec (225 kt)

Collective Control System

Servo rate limits (no load)	5.62 in./sec
Gyro and control system effective mass	8.7 slugs

Directional Control System

Pedal travel	5.9 in. (approx)
Trim authority:	
Positive blade	142 percent
Negative blade	89 percent
Servo rate limits (no load)	3.75 in./sec

MAIN ROTOR

13. The four-bladed main rotor features blade articulation about the feathering axes only, hence is referred to as "rigid." The hub consists of fixed and movable portions. The fixed hub is attached solidly to the rotor mast while the four movable hub elements provide transition structure to the blade roots. Blade feathering motion is provided by a "door hinge" between the fixed and movable hub sections. Blade flapping and lead-lag motion are resisted by structural deflection of the blades and hub. The rotor blade cross section is of constant chord and varying thickness and section. Basically, the root section is a droop-nose modification of a NACA 23012 airfoil, while the tip section is a modified NACA 23006 airfoil.

14. The main rotor is controlled by an externally-mounted gyro which is mechanically in series between the rotor blades and the swashplate (the plane of the swashplate is identical to the plane of the gyro). The gyro is gimballed to the rotor mast. hence free to establish its own plane in space. The main rotor blade is swept forward of its reference radial by means of offset blade root attachment bolts; thus, when the blade flaps vertically a feathering moment is felt at the pitch arm. This moment is applied to the gyro through the pitch arm/pitch link. Rotor blade feathering is controlled by gyro tilt; this tilt (plane in space) is determined by the balance of moments caused by the pilot's control inputs, blade feathering moments, and gyro precession rates.

15. This arrangement is designated by LCC as a gyro-controlled rotor, and performs two functions; aircraft stability and rotor loads alleviation. The pilot flies the aircraft by his boosted inputs to the control gyro, which then precesses due to the gyro moment imbalance and inputs cyclic blade angle changes to the main rotor. When the main rotor is displaced by an external disturbance (such as a vertical gust) and flaps upward, the gyro imbalance due to the feathering moment signal will cause the gyro to precess, changing main rotor blade feathering to "wash out" the gust effects. By this stabilization of the rotor, the control gyro alleviates the rotor loads due to the gust. In addition, the gyro limits the rotor loads due to sudden abrupt cyclic inputs by the pilot, since rate of change of cyclic blade angle is limited by the gyro precessional rate due to the pilot input moment.

16. Because the mechanism provided to sense blade flapping stresses utilizes pitching moment, a number of extraneous signals are also fed to the gyro. These include the product of blade inplane moments acting through the effective blade droop angle, feathering moments due to C_{mo} and $C_{m\alpha}$ of the rotor blade, pitch damping, feathering inertia, and door-hinge friction. Considerable effort has been spent during the contractor development program to optimize the rotor geometry to account for all these phenomena and related rotor response.

17. Prior to APE I.3, the main rotor droop was increased from 2 degrees, 20 minutes to 3 degrees, 10 minutes to provide rotor stability at increased gross weights. Between APE I.3A and APE I.3B the main rotor tip weights were removed, then reinstalled for rotor stability reasons. Principal main rotor characteristics are tabulated below:

Blade designation with tip weight	1019765
Fixed hub designation, soft inplane	1019772

Movable hub designation	1018578
Pitch arm designation, "zero δ_3"	1018569
Hub location (contact surface of bottom of fixed hub with shaft flange gasket):	
Fuselage station	300.0
Water line	165.3
Built-in coning	2 deg
Shaft incidence	Zero deg
Number of blades	4
Airfoil section.	
Root	NACA (4.6) 3012 modified
Tip	NACA (0.6) 3006 modified
Radius	25.617 ft
Chord (all computations based on c = 28 in. (theoretical)):	
Rotor station 79.12	27.50 in.
Rotor station 140.0 (linear taper)	27.60 in.
Rotor station 170.0 (between stations)	27.66 in.
Rotor station 302.4	27.89 in.
Rotor station 302.4 to tip	27.89 in.
Droop	2 deg, 20 min (APE I.1, APE I.2, RDAT I) and 3 deg, 10 min (APE I.3)
Sweep, forward	4 deg, 00 min

Disc area, πR^2	2062 ft^2
Blade area, bcR	239.1 ft^2
Solidity, σ = bc/πR	0.1159
Geometric twist, θ_1, from center of rotation to rotor station 302.4	-5 deg
Tab location, fuselage station at tab centerline	264.0
Tab size, equivalent	28.1 in. x 2 in.
Collective pitch range, θ_0	Zero deg, 30 min to 18 deg, 30 min
Normal rotor speed	246 rpm
Angular velocity	25.76 rad/sec
Normal tip speed	660 ft/sec
Blade inertia about 1/4 chord	12,295.4 lb-in.^{2}
Increment of blade inertia due to:	
Discrete weights	23 lb-in.^{2}
Polar moment of inertia	9748 slug-ft^2
Dynamic system equivalent polar moment of inertia includes main rotor, tail rotor, and propeller	10,742 slug-ft^2

TAIL ROTOR

18. A four-bladed teetering antitorque rotor is mounted at the tip of the left horizontal stabilizer. The blades have a constant 14-inch chord with a slab-sided droop-nosed cross section. The thrust is inboard. Direction of rotation is clockwise when viewed from the left side of the ship looking inboard. Principal tail rotor characteristics are tabulated below:

Blade designation	1C19380
Hub designation	1019381

Hub location (teeter center):	
Fuselage station	658.5
Water line	114.5
Buttline	72.0 left
Built-in coning	Zero deg
Number of blades	4
Airfoil section	NACA (.675) 300 (5.89) modified
Radius	5 ft
Chord	1.167 ft
Disc area	78.5 ft^2
Theoretical blade area, bcR	23.3 ft^2
Solidity, $\sigma = bc/\pi R$	0.297
Twist, θ_1	Zero deg
Pitch range	-10 deg to +22.5 deg
Maximum allowable tilt	15 deg
Delta-three	37.5 deg
Normal rotor speed	1238 rpm
Angular velocity	129.6 rad/sec
Normal tip speed	648 ft/sec
Tail rotor moment arm, l_{tr}	29.88 ft
Polar moment of inertia	12.6 slug-ft^2

PROPELLER

19. Longitudinal thrust is provided by a Hamilton Standard pusher propeller mounted at the rear of the fuselage. The propeller is capable of providing forward and reverse thrust. The direction of rotation is counterclockwise when viewed from behind the aircraft looking forward.

20. The pilot controls the propeller by using a twist grip located on the collective lever. The twist grip rotates 140 degrees corresponding to 58 degrees of blade angle change from -17.2 degrees to +40.8 degrees. The relationship is nonlinear, in that increased twist grip rotation is required at large blade angles (*ie*, 3:1 from 35 to 40 degrees of Beta versus 2.1 from -10 to -5 degrees of Beta). On aircraft S/N 66-8834, the negative Beta was restricted to -5 degrees because of instrumentation.

21. An automatic system (Delta Beta) senses main rotor shaft torque and load factor to provide a reduction of propeller pitch to 18 degrees to minimize rotor speed decay in case of an engine failure or a power chop. Principal propeller characteristics are tabulated below:

Propeller designation	Hamilton Standard 1311 GB 30/11FA 10A4-0
Hub location:	
Fuselage station	675.7
Water line	114.5
Shaft incidence	Zero deg
Number of blades	3
Radius	5 ft
Activity factor per blade	142
Integrated design lift coefficient	0.411
Pitch range (physical limits, at blade station 42)	-17.2 to 40.8 deg
Pitch range (flight test limits, at blade station 42 with oil damping and counterweights installed for failure mode):	
Aircraft S/N 66-8831	-12 to 40.8 deg
Aircraft S/N 66-8834	-5 to 40.8 deg

Direction of rotation, viewed from rear	Counterclockwise
Normal propeller speed	1717 rpm
Angular velocity	179.8 rad/sec
Normal tip speed	899 ft/sec
Polar moment of inertia	13.98 slug-ft^2

WING

22. The wing is of trapezoidal planform and is mounted on the sponsons with the 0.25 mean aerodynamic chord (MAC) located at FS 308.2. Originally the section was a four-digit NACA airfoil, but early in the contractor development program additional wing area was added. This was accomplished by extending the wing trailing edge and providing transition fairings in the former aft wing region. Compensation for rolling moment due to propeller torque is provided by an increased incidence angle on the right wing. Prior to APE I.3A, the right wing incidence was further changed to provide more beneficial main rotor lateral loading and reduced aircraft vibration. Prior to APE I.3B, detuning weights were added to the right wing to reduce local vibration. Principal wing characteristics are tabulated below:

Wing designation	1016648
Airfoil:	
Root, buttline zero	12 percent
Tip, buttline 160.2	8 percent
Area	195 ft^2
Span	26.7 ft
Aspect ratio	3.66
Mean aerodynamic chord	7.6 ft
Fuselage station at 1/4 MAC	308.2
Taper	0.50
Dihedral	5 deg

Incidence:	
Left wing	11 deg, 52 min
Right wing	12 deg, 58 min
Trailing edge deflection, right wing	3 deg, down (APE I.1, APE I.2) and 1 deg, down (APE I.3, RDAT I)
Twist:	
Left wing	-3 deg, 6 min
Right wing	-3 deg, 2 min

HORIZONTAL STABILIZER

23. The horizontal stabilizer is mounted at the aft end of the fuselage and has a basically trapezoidal planform. The cross section of the stabilizer is a modified symmetric airfoil. The right stabilizer has tapering thickness. The left stabilizer is truncated in the chordwise direction, resulting in a bobtail appearance. Detuning weights are provided to reduce local vibration. Prior to APE I.3A, these weights were removed. Also prior to APE I.3A, the horizontal stabilizer incidence was modified to reduce aircraft vibration. Principal horizontal stabilizer characteristics are tabulated below:

Horizontal stabilizer designation:	
Left side, Phase II reverse rotation	1019548
Right side	1000667
Airfoil:	
Right panel:	
Root, buttline zero	NACA 0018 modified
Tip, buttline 65.0	NACA 0012 modified
Left panel (highly modified, bobtailed):	NACA 0018

Area:	
Left side	16.25 ft^2
Right side	15.58 ft^2
Total	31.83 ft^2
Span	10.83 ft
Aspect ratio	3.68
Mean aerodynamic chord:	
Left side	36.84 in.
Right side	35.40 in.
Average	36.12 in.
Fuselage station of 1/4 MAC:	
Left side	637.38
Right side	636.98
Average	637.18
Taper:	
Left side	0.583
Right side	0.568
Average	0.576
Dihedral	Zero deg
Incidence	2 deg
Twist	Zero deg
Deflection of right-hand trailing edge	2.8 deg, down

VERTICAL STABILIZER

24. The vertical stabilizer is mounted ventrally under the aft end of the fuselage. The cross section is an 18-percent symmetrical airfoil with no incidence relative

to the fuselage centerline. The tail wheel is mounted within the lower end of the stabilizer and is retracted up into the stabilizer in flight. Principal characteristics of the vertical stabilizer are tabulated below:

Vertical stabilizer designation, Phase II	1000594
Airfoil section:	
Root, water line 114.5	NACA 0018 modified
Tip, water line 37.6	NACA 0018 modified
Area, between water line 37.6 and water line 114.5	24.6 ft^2
Span	6.41 ft
Aspect ratio	1.67
Mean aerodynamic chord	3.92 ft
Location of 1/4 MAC:	
Fuselage station	620.3
Water line	79.4
Taper	0.587
Incidence	Zero deg

APPENDIX C. SAFETY-OF-FLIGHT RELEASES

1. This appendix contains the safety-of-flight releases, amendments, and flight envelopes for APE I.1, APE I.2, APE I.3, and the USAASTA portion of RDAT I.

2. Figures 1 and 2 referenced in the APE I.1 safety-of-flight release have been combined with and are presented with the APE I.2 safety-of-flight release.

AMSAV-R-F

SUBJECT: APE I.1 Safety of Flight Release

Commanding Officer
US Army Aviation Systems
Test Activity
ATTN: SAVTE-P
Edwards AFB, California 93523

1. This letter constitutes a safety of flight release for day V.P.R. flight of AH-56A S/N 66-8834 without wing stores for the conduct of Army Preliminary Evaluation I.1.

2. This flight release is contingent upon the following:

a. The airworthiness of all onboard flight test equipment and instrumentation being assured by a safety inspection performed by USAASTA personnel.

b. An inspection of cockpit displays being accomplished by USAASTA personnel to assure these displays are properly marked to reflect the flight and operating limitations specified in this flight release.

c. The flight control systems being rigged in accordance with applicable drawings and specifications.

3. The authorized flight envelope is as described below.

a. Airspeed Limitations.

(1) Forward Flight. The maximum authorized forward flight speed is 190 knots calibrated airspeed (airspeed indicator red line).

(2) Landing Gear Extended. The maximum authorized flight speed with the gear extended is 130 knots calibrated airspeed.

(3) Butterfly Canopy (Forward and/or Aft). The canopy open (forward and/or aft) condition is authorized only for ground conditions, rotor stationary, and winds of 45 knots or less.

(4) Taxi

(a) Tail Wheel Unlocked. The maximum authorized taxi speed with the tail wheel unlocked is 20 knots.

(b) Tail Wheel Locked. The maximum authorized taxi speed with the tail wheel locked is 70 knots calibrated airspeed.

(5) Sideward Flight (or hovering in winds). The maximum authorized sideward flight speed is 35 knots (See Figure 2, sideslipsenvelope). Prop blade angle shall be set at -2.2°.

(6) Rearward Flight (or hovering in winds). The maximum authorized rearward flight speed is 30 knots. Prop blade angle shall be set at -2.2°.

b. Collective Blade Angle. Collective/main rotor swashplate position is sensed and presented on a cockpit display in degrees. The maximum authorized cockpit displayed collective angles are:

Calibrated Airspeed	Collective Angle
Hover to 80 knots	As required
80 knots to 120 knots	7°
120 knots to 190 knots	5°

c. Bank Angle Limitations. The maximum authorized bank angles as a function of calibrated airspeed are shown in Figure 1. (Incl 1)

d. Sideslip Envelope. The maximum authorized sideslip as a function of calibrated airspeed is shown in Figure 2. (Incl 2)

e. Descents. The maximum authorized rate of descent is 6000 feet per minute. Flight path (dive) angle is limited to 20 degrees or less and a propeller beta angle of 8 degrees or greater.

f. Control Input Limits. Abrupt pedal inputs in forward flight shall not exceed ±1 inch from trim or result in sideslip angles greater than that authorized by the sideslip-airspeed envelope shown in Figure 2.

g. Yaw Rate Limits. Yaw rates during hover turns on a spot shall not exceed 45 deg/sec right or left.

h. Load Factor. The authorized load factor-airspeed evelope is shown in Figure 3. (Incl 3)

i. Altitude Limits. Flight above 6000 ft. density altitude is prohibited.

j. Weight and C.G. Limits. The maximum allowable take-off gross weight is 18,800 pounds. The aircraft shall be configured to the mid-center of gravity (c.g.) with a permissible c.g. travel from station 299 to station 301.

k. Rotor Speed Limits. Steady state maneuvers shall be conducted at 100% N_R and transient maneuvers may vary from 90% to 105% N_R.

4. The engine, transmission, hydraulic system, and APU limitations and associated instrument markings are as follows:

LIMITATIONS	INSTRUMENT MARKING
a. Turbine Inlet Temperature	
200°C minimum at flight idle.	red line
(Note: Under normal conditions the flight idle TIT will be in the 400-440°C range. TIT values on the order of 200°C can be obtained at flight idle under the extremes of high altitudes and low temperature (-65°F))	
200 - 671°C normal operation	green base
566°C max allowable on start up to 50% Hg	(no mark)
649°C max allowable on start up to idle	(no mark)
671°C steady state (max continuous)	red line
671 to 707°C time limited - 30 min.	yellow band
707 to 727°C time limited - 10 min.	yellow band

LIMITATIONS	INSTRUMENT MARKING
Turbine Inlet Temperature continued	
727-743°C the maximum allowable transient may exceed 727°C for not more than 60 seconds, but must not exceed 743°C	yellow band
743°C inspection limitation	red line
b. Gas Generator RPM	
58% idle (minimum)	red line
72% idle (maximum)	(no mark)
63 to 100% normal (run) operating range	green band
100 to 101.5% for 10 seconds	yellow band
101.5% maximum	red line
c. Power Turbine RPM	
95% (Power on) minimum	red line
95% to 105% normal operating range	green band
105% maximum (power on)	red line
overspeed cutoff - 113%	(no mark)
d. Main Rotor RPM	
90% minimum power off (95% minimum power on)	red line
90 to 105% normal operating range	green band
105% maximum power off	red line
e. Engine Torque	
0 - 100% normal operating range	(no band)
100% (3435 SHP at 100% RPM)	yellow band
114% (3925 SHP at 100% RPM)	red line

LIMITATIONS	INSTRUMENT MARKING
f. Fuel Flow Indicator	
103 pph minimum idle	red line
350 pph maximum idle	yellow line
350 pph to 1720 pph operating range	green band
1720 pph maximum	red line
g. Fuel Quantity Indicator	
0 to 390 lb low fuel level warning	yellow band
h. Engine Oil Temperature	
0°C minimum	red line
0 to 107°C normal operating range	green band
107°C maximum	red line
107°C to 150°C for 30 minutes - emergency only above 150°C see Para. 6, Emergency Procedures.	(no mark)
i. Transmission Oil Temperature	
-30°C minimum	red line
0 to 113°C normal operating range	green band
113°C maximum	red line
113°C to 130°C for 30 minutes and 130°C to 140°C for 10 minutes (For emergency only and with power level equal to power for level flight at 90 to 100 KTS)	(no marks)
j. Engine Oil Pressure	
10 psi minimum	red line
10 psi to 45 psi idle	(no mark)

LIMITATIONS	INSTRUMENT MARKING
Engine Oil Pressure continued	
10 psi to 75 psi	green band
75 psi*	red line
k. Transmission Oil Pressure	
80 psi minimum	red line
80 to 110 psi normal operating range**	green band
110 psi maximum	red line
l. Hydraulic System Pressure	
0 to 2800 psi	yellow band
2800 to 3400 psi normal operating range	green band
3400 psi maximum	red line
Bypass mode -2 minute maximum	(no mark)
m. APU RPM	
95% minimum	red line
95 to 100% normal operating range	green band
100 to 110%	yellow band
110% maximum	red line
n. APU Exhaust Gas Temperature	
50 to 650°C normal operating range	green band
650°C maximum continuous	red line

*When starting in cold weather, oil pressures greater than 100 psi can occur before oil temperature stabilizes (within three minutes)
**Special conditions apply to operation with transmission oil pressure below 85 psi, see Para. 6, Emergency Procedures.

LIMITATIONS	INSTRUMENT MARKING
APU Exhaust Gas Temperature continued	
650 to 732°C	yellow band
732°C time limit - 10 sec	red line
o. Prop Gearbox Oil Temperature	
121°C maximum	red line

5. Ejection seat restrictions:

The ejection seat was installed in Aircraft 66-8834 for the purpose of providing emergency egress for the contractor pilot during envelope expansion flights. The ejection seat has not been qualified in this aircraft and therefore the use of the ejection seat during the APE I.1 evaluation will be at the discretion of the aircraft Commander. The interdepartmental communication from Mr. D.R. Segner, subject: AH-56 Ejection Seat Qualification, dated 20 May 1970, contains the controlling guidelines for the use of the ejection seat.

6. Emergency Procedures:

a. Checklist Emergency Procedures: The emergency procedures detailed in POMM 55-1520-22-10 CL, (January 1971), Operator's and Crewmember's Checklist, for aircraft serial no. 66-8834 shall be followed with special emphasis on the following:

(1) Prop System Control Failure - page E9.

(2) Proximity Device Warning - page E20. This system has been removed.

(3) Stick Centering Malfunction/Failure - pages E26 and E27.

(4) N_f Control Failure - page E28.

b. Additional Emergency Procedures. The following emergency procedures not included in the pilot's checklist should be followed:

(1) In flight emergency egress from the aft cockpit should be out the righthand side to avoid possible contact with the tail rotor.

(2) Low transmission oil pressure (below 85 psi) does not require an immediate precautionary landing. Flight may be continued for a maximum of 30 min. at 90-100 KIAS if the following transmission oil pressure vs. inlet temperature conditions are not exceeded:

80 psi @ 80°C

77 psi @ 85°C

73 psi @ 90°C

70 psi @ 95°C

(3) Engine oil overtemperature due to oil cooler fan failure does not require an immediate precautionary landing if engine oil pressure is above 40 psi and an airspeed of 130 KIAS can be maintained.

7. Cautions and Warnings:

a. Caution. Blade moment stall has been encountered at conditions of 7° collective pitch blade angle, as presented by the cockpit display, 120 knots calibrated airspeed, and load factors of 1.8 or greater. This condition is characterized by right roll and pitch up. Recovery techniques shall be consistent with the procedures demonstrated to USAASTA pilots by Lockheed during the pilot training.

b. Caution. During Pre-Engine Start System Checks insure that the RPM Set Switch (N_f Beeper) has been set in the DECR position for a minimum of five seconds.

c. Caution. Do not apply rotor brake with engine running. Apply rotor brake only below 40% N_R with engine off and TIT below 320°C. Rotor brake may be applied before engine start but must be released at ground idle. Do not attempt to keep rotor brake on beyond ground idle when running up.

d. Caution. Because of fuel control NASH actuator unreliability the engine twist grip is not to be used for practice autorotations.

e. Warning. Do not start APU with rotor turning and a known or suspected No. 1 hydraulic system malfunction at any time.

f. Warning. Landing roll decelerations must be accomplished using reverse propeller thrust and main gear braking only. Aft cyclic inputs during ground operation can overstress main rotor control components or airframe structure.

g. Warning. Avoid operation at 40°F or below with visible moisture present.

8. Limited Life Parts:

a. The maximum allowable operating times (MAOT) for fatigue critical component parts are as listed in Table 1. (Inc

b. USAASTA personnel shall assure that the special inspections indicated under the S.I. column of Table 1 are performed at the intervals specified in document IDC 87-71-045, "Tracking of Finite Life and TBO Items", date 20 Jan 1971.

c. Special Inspection 25.1 and 25.2 shall, in addition to the normal inspection intervals, be performed immediately prior to APE I.1.

9. Propeller Blade Angle Limitations:

a. Maximum allowable propeller blade angle during acceleration through transition for V.T.O.L. take-off is +25°.

b. Maximum allowable propeller blade angle during climb below 100 knots calibrated airspeed is +30°.

c. Propeller blade angles in excess of +40.8° are prohibited.

d. Propeller blade angles less than -5° are prohibited.

10. Safe Take-Off Corridor: Note: Contractor has demonstrated throttle chops from hover to 60 knots at 7 to 10 feet wheel heights.

FOR THE COMMANDER:

CHARLES C. CRAWFORD, JR.
Director of Flight Standards
and Qualification

AMSAV-R-F
SUBJECT: APE I.1 Safety of Flight Release

Copy Furnished:
Advanced Aerial Weapons Systems
Project Managers

Lockheed - California Company

APE I.1 Flight Release
AH-56 SN 66-8834
AMSAV-R-F

TABLE I

ITEM	PART NUMBER	COMPONENT	MAOT	S.I.
1.	1019765-301	M/R Blade Assy	210	15.4 180.4
2.	1018578-303	M/R Movable Hub Assy	380	30.9 60.2 180.11
3.	2602853	M/R Tension Torsion Pak	200	
4.	LS10066-1	M/R T-T Pak Pin (8)	3600	15.8 180.2
5.	LS10065-3	M/R Hinge Bolt	250	30.4
6.	740290-107	M/R Hinge Bearing	250	30.4
7.	1019772-301	M/R Fixed Hub Assy	340	30.8
8.	1010460-305	M/R Gyro Dome	730	60.7
9.	1013452-301	M/R Gyro Drive Assy	950	
10.	1013447-301	M/R Gyro Hub	1970	
11.	1019896-101	M/R Gyro Arms	2700	
12.	1018569-319	M/R Pitch Arm	65	2.3 30.7 50.2
13.	1010335-325	M/R Servo Package	350	
14.	1019881-105	M/R Coll Bellcrank	260	
15.	1019989-101	M/R Coll Link	200	
16.	1013450-307	Rotating Push Rod	900	
17.	1009430-305	Rotating Swashplate Assy	900	
18.	1011572-305	Non-Rotating Swashplate	1710	
19.	1013916-101	Swashplate Ball Spline	310	
20.	1001981-107	Ball Spline Housing	210	
21.	1013432-305	M/R Pitch Link	1150	60.1
22.	100723A3	M/R Collective Dampers	450	
23.	673171-101	M/R Cyclic Dampers	480	
24.	1000156-177	Main Transmission (Sprag)	150*	2.1 30.6
25.	1000211-103	M/R Mast (Trans. 1010)	1340	
26.	1020252-105	Sprag Clutch	50	25.1
27.	670869-103	A.P.U.	1800	
28.	670825-103	Lube Pump	150	
29.	671766-301	A.P.U. Clutch	1200	
30.	740530-105	A.P.U. Drive Shaft	1200	
31.	T-64-GE-16	Engine	150	25.2
32.	1020251-111	Engine Torqueshaft	50	25.2
33.	671221-301	Nash Actuator Fuel Control	1200	
34.	1020263-101	Engine Speed Control	50	25.1
35.	1002063-101	Propeller Drive Shaft	1200	
36.	738700-1	Propeller Gearbox	200	
37.	11FA10B8-0	Propeller Blades	1200	

APE I.1 Flight Release
AH-56 SN 66-8834
27 Jan 1971
AMSAV-R-F

ITEM	PART NUMBER	COMPONENT	MAOT	S.I.
38.	738620	Propeller Actuator	1200	
39.	1015146-301	T/R Drive Shaft Assy	230	
40.	1007925-303	T/R Gimbal Ring Assy	140	
41.	1008840-303	T/R Spindle Mechanism	300	
42.	1009256-301	T/R Hub Mech. Assy	1310	
43.	1019382-301	T/R Spider - Pitch Control	300	
44.	1019370-301	T/R Blade Assy	410	
45.	1009327-301	T/R Feather Arm	1200	
46.	1008898-305	T/R Spline Shaft	280	Inwork
47.	1001707-301	M/R Blade Attach Bold, Fwd.	470	180.1
48.	1018922-103	M/R Blade Attach Bolt, Aft	470	
49.	1020319-101	M/R Collective Clevis	2380	
50.	101492-301	M/R Cyclic Clevis	2150	
51.	1014055-309	Pushrod Assy-Non Rotating	2530	
52.	1014055-310	Pushrod Assy-Non Rotating	2530	
53.	1013915-303	Pushrod Assy-Non Rotating	2530	
54.	1020312-101	Cone	1710	
55.	1009989-301	Retainer	2530	
56.	1020[illegible]83-101	Link Assy	1710	
57.	1015595-313	Link Assy	1710	
58.	740396-301	T/R Tension Torsion Pack	1310	
59.	1010445-301	T/R T.T Pack Pin	1310	
60.	1009338-301	T/R T.T Pack Pin	1310	
61.	1007712-301	T/R Spindle Support Housing	180	
62.	1008361-303	T/R Drive Spindle	140	
63.	740599-301	T/R Pitch Link	900	
64.	1008899-301	T/R Bell Crank	340	
65.	1007460-301	Bellcrank, Direct Control	1980	
66.	1007462-301	Support, Direct Control	1980	
67.	1007464-301	Link Assy, Direct. Control	1980	
68.	1009046-101	Rod Assy, Direct. Control	1980	
69.	1001690-115	Directional Servo Assy	1200	
70.	1000663-101	Fitting, Front Beam	450	60.8 60.10
71.	1000665-101	Fitting, Rear Beam	450	60.8 60.8 60.10

Page 2 of Table 1.

APE I.1 Flight Release
AH-56 SN 66-8834
27 Jan 1971
AMSAV-R-F

ITEM	PART NUMBER	COMPONENT	MAOT	S.I.
72.	740882-301	T/R Feather Bearing		5.1**
73.	740597-303	T/R Drive Spindle Bearing		15.6**
74.	740718-301	T/R Gimbal Bearing		15.6**
75.	670215-105	Oil Cooler Fan - Vickers		**, ***
76.	740718-301	Oil Cooler Fan - Task		**, ***

*Exclusive of sprag clutch.

**Special inspection and replacement procedure based on cumulative wear.

***Visual inspection for leakage prior to and after each flight.

Page 3 of Table 1.

NNNNPTTUZYUW RUWTFFA3828 Ø492227-UUUU--RUWJBDA.

ZNR UUUUU

P 1821ØØZ FEB 71

FM CG USAAVSCOM ST LOUIS MO

TO CO USAASTA EDWARDS AFB CA

BT

UNCLAS

AMSAV-R-F

ACTION FOR: SAVTE-P

SUBJECT: APE 1.1 SAFETY OF FLIGHT RELEASE

IN REPLY REFER TO: AMSAV-R-FT Ø2-13

A. REFERENCE IS MADE TO LETTER FROM AMSAV-R-F TO SAVTE-P, SUBJECT AS ABOVE, 29 JAN 71.

PARAGRAPH 3.H OF ABOVE REFERENCE IS HEREBY MODIFIED AS FOLLOWS:

LOAD FACTOR. THE AUTHORIZED LOAD FACTOR AIRSPEED ENVELOPE IS SHOWN IN FIGURE 3 FOR AL CONDITIONS EXCEPT RIGHT WINDUP TURNS WHICH SHALL NOT EXCEED 1.5G'S BETWEEN 8Ø AND 155 KNOTS THEREBY DECREASING LINERLY TO 1.2G'S AT 19Ø KNOTS.

BT

#3828

MNNNPTTUZYUW RUWTFFA4412 Ø551552-UUUU--RUWJBDA.

Z NR UUUUU

P 2215ØØ Feb 71

FM CG USAAVSCOM STL MO

TO CO USAASTA EDWARDS AFB CALIF

BT

UNCLAS

AMSAV-R-F

FOR: SAVTE-P

SUBJ: APE 1.1 SAFETY OF FLIGHT RELEASE

IN REPLY REFER TO AMSAV-R-F Ø2-18

A. REFERENCE IS MADE TO LETTER FROM SAMSAV-R-F TO SAVTE-P, SUBJECT AS ABOVE, 29 JAN 71.

THE REFERENCE SAFETY OF FLIGHT RELEASE IS HEREBY EXPANDED TO INCLUDE THE FOLLOWING LIMITATIONS:

A. AUTOROTATIONAL ENTRY DURING CLIMBS ARE RESTRICTED TO CLIMB RATES OF APPROXIMATELY 5ØØ FT/MIN.

B. AUTOROTATIONAL ENTRY CONTROL DELAYS SHALL BE HELD TO APPROXIMATELY ONE (1) SECOND, UNLESS PILOT CUES ARE OF SUFFICIENTLY LOW MAGNITUDE TO ALLOW DELAYS UP TO A MAXIMUM OF TWO (2) SECONDS.

BT

#4412

DEPARTMENT OF THE ARMY
US ARMY AVIATION SYSTEMS COMMAND
PO Box 209, St. Louis, MO 63166

AMSAV-R-F

19 APR 1971

SUBJECT: APE 1.2 Safety of Flight Release

Commanding Officer
U.S. Army Aviation
Systems Test Activity
ATTN: SAVTE-P

1. This letter constitutes a safety of flight release for day V.F.R. flight of AH-56A S/N 66-8834 for the conduct of Army Preliminary Evaluation 1.2.

2. This flight release is contingent upon the following:

a. The airworthiness of all onboard flight test equipment and instrumentation being assured by a safety inspection performed by USAASTA personnel.

b. The flight control systems being rigged in accordance with applicable drawings and specifications.

c. Constant voice communication will be maintained between the Telemetry van and the test vehicle.

d. The mechanical emergency stores jettison system must be functionally tested by the contractor prior to APE 1.2 flight testing.

e. The emergency stores jettison handle must be painted and identified per MIL-M-18012 for immediate action controls.

3. The authorized flight envelope is as described below.

a. Airspeed Limitations.

(1) Forward Flight . . . The maximum authorized forward flight speed is 160 knots calibrated airspeed.

(2) Landing Gear Extended . . . The maximum authorized flight speed for normal landing gear extension (or with the landing gear extended) is 130 knots calibrated airspeed.

(3) Butterfly Canopy (Forward and/or Aft) . . . The canopy open (forward and/or aft) condition is authorized only for ground conditions, rotor stationary, and winds of 45 knots or less.

(4) Taxi, Takeoff, and Landing . . .

(a) Tail Wheel Unlocked . . . The maximum authorized taxi speed with the tail wheel unlocked is 20 knots.

(b) Tail Wheel Locked . . . The maximum authorized taxi speed with the tail wheel locked is 70 knots calibrated airspeed.

(5) Sideward and Rearward Flight . . . These maneuvers are prohibited.

(6) Takeoffs . . . Running takeoffs only are authorized and the minimum lift-off airspeed shall be in accordance with Figure 6, Incl 7.

(7) Landings . . . Run-on landings only are authorized and maximum airspeed at touchdown is limited to 70 knots calibrated airspeed.

(8) Autorotative Descent . . . Stabilized autorotative descent airspeed shall be limited to 85 to 95 knots calibrated airspeed.

b. Collective Blade Angle . . . Collective/main rotor swashplate position is sensed and presented on a cockpit display in degrees. The authorized cockpit displayed collective angles as a function of airspeed are shown in Figure 4, Incl 5.

c. Bank Angle Limitations . . . The maximum authorized bank angles as a function of calibrated airspeed are shown in Figure 1, Incl 2.

d. Sideslip Envelope . . . The maximum authorized sideslip as a function of calibrated airspeed is shown in Figure 2, Incl 3.

e. Descents . . . The maximum authorized rate of descent is 6000 feet per minute. Flight path (dive) angle is limited to a maximum of 20 degrees with a minimum propeller beta angle of +8 degrees except during landing.

f. Hovering . . . Hovering flight is prohibited.

g. Practice/Intentional Autorotations . . . These maneuvers are prohibited.

h. Control Input Limits, Directional . . . Abrupt pedal inputs in forward flight shall not exceed ±1 inch from trim or result in sideslip angles greater than that authorized by the sideslip-airspeed envelope shown in Figure 2, Incl 3.

i. Control Input Limits, Cyclic . . .

(1) Ground 100% N_R . . . Cyclic control inputs shall be limited to ±2 inches.

(2) Cyclic stirs . . . cyclic stirs at rates greater than one cycle in two seconds (0.5Hz) are prohibited.

j. Load Factor . . . The authorized load factor airspeed envelope is shown in Figure 3, Incl 4.

k. Altitude Limits . . . Flight above 6000 ft. density altitude is prohibited.

l. Weight and C.G. Limits . . . The maximum allowable takeoff gross weight is 18,500 pounds. The aircraft shall be configured to the mid-center of gravity (c.g.) with a permissible longitudinal c.g. travel from station 299 to station 301, and a lateral c.g. travel of ±1 inch.

m. Rotor Speed Limits . . .

(1) Steady State Maneuvers, power on or power off - 98% to 105% N_R.

(2) Transient Maneuvers, power on - 95% to 105% N_R.

n. Rotor Start Stop Limits . . . The rotor shall not be started or stopped in winds in excess of 20 knots.

o. Touch Down Sink Rates . . . Touch down sink rate shall not exceed 9.5 feet per second.

p. Wind Limits . . . Flight operations shall not be conducted in winds in excess of 20 knots.

q. External Stores Jettison . . . In the event of an emergency, TOW missile pods and/or XM-159C rocket launchers may be jettisoned in the speed range of 0 to 100 knots calibrated from a level unaccelerated flight condition. Said jettison may be accomplished with any combination of full, partially loaded or empty TOW pods and XM-159C rocket launchers. This limitation covers TOW pods mounted on the B.L. 68 pylon and XM-159C rocket pods mounted on either the B.L. 68 pylon and/or B.L. 117 pylon.

4. The engine, transmission, hydraulic system, and APU limitations and associated instrument markings are as follows:

LIMITATIONS	INSTRUMENT MARKING
a. Turbine Inlet Temperature	
200°C minimum at flight idle	red line
(Note: Under normal conditions the flight idle TIT will be in the 400-440°C range. TIT values on the order of 200°C can be obtained at flight idle under the extremes of high altitudes and low temperature (-65°F)	
200 - 671°C normal operation	green band
566°C max allowable on start up to 50% N_g	(no mark)
649°C max allowable on start up to idle	(no mark)
671°C maximum continuous limit	red line
671 to 707°C time limited - 30 min	yellow band
707 to 727°C time limited - 10 min	yellow band
727 - 743°C time limited - 60 seconds	yellow band
743°C inspection limitation	red line
b. Gas Generator RPM	
58% idle (minimum)	red line
72% idle (maximum)	(no mark)

LIMITATIONS	INSTRUMENT MARKING
Gas Generator RPM continued	
63 to 100% normal (run) operating range	green band
100% maximum continuous	(no mark)
100 to 101.5% time limited to 10 seconds	yellow band
101.5% inspection limitation	red line
c. Power Turbine RPM	
98% (power on) minimum	red line
98% to 105% normal operating range	green range
105% maximum (power on)	red line
overspeed cutoff - 113%	(no mark)
d. Main Rotor RPM	
90% minimum power off (98% minimum power on)	red line
98 to 105% normal operating range	green band
105% maximum power on/off	red line
e. Engine Torque	
0 - 114% normal operating range	(no band)
114% maximum continuous	red line
f. Fuel Quantity Indicator	
0 to 390 lb low fuel level warning	yellow band
g. Engine Oil Temperature	
0°C minimum (power on)	red line

AMSAV-R-F 19 APR 1971

SUBJECT: APE 1.2 Safety of Flight Release

LIMITATIONS	INSTRUMENT MARKING
Engine Oil Temperature continued	
0 to 107°C normal operating range	green band
107°C maximum continuous	red line
107°C to 150°C for 30 minutes - emergency only above 150°C see Para 6, Emergency Procedures	(no mark)
b. Transmission Oil Temperature	
-30°C minimum	red line
0 to 113°C normal operating range	green band
113°C maximum continuous	red line
113°C to 130°C for 30 minutes and 130°C to 140°C for 10 minutes (For emergency only and with power level equal to power for level flight at 90 to 100 KIAS)	(no mark)
i. Engine Oil Pressure	
10 psi minimum	red line
10 psi to 45 psi idle	(no mark)
10 psi to 75 psi	green band
50 psi (minimum at 95% N_g)*	(no mark)
j. Transmission Oil Pressure	
80 psi minimum	red line
80 to 110 psi normal operating range**	green band
110 psi maximum	red line

* When starting in cold weather .1 pressures greater than 100 psi can occur before oil temperatur. abilized (within three minutes).

** Special conditions apply to operation with transmission oil pressure below 80 psi, see Para. 6, Emergency Procedures.

LIMITATIONS	INSTRUMENT MARKING
k. Hydraulic System Pressure	
0 to 2800 psi	yellow band
2800 to 3400 psi normal operating range	green band
3400 psi maximum	red line
Bypass mode -2 minute maximum	(no mark)
l. APU RPM	
95% minimum	red line
95 tc 100% normal operating range	green band
100 to 110%	yellow band
110% maximum	red line
m. APU Exhaust Gas Temperature	
50 to 650°C normal operating range	green band
650°C maximum continuous	red line
650 to 732°C	yellow band
732°C time limit - 10 sec	red line
n. Prop Gearbox Oil Temperature	
121°C maximum	red line

5. Ejection Seat Restrictions:

The ejection seat was installed in Aircraft 66-8834 for the purpose of providing emergency egress for the contractor pilot during envelope expansion flights. The ejection seat has not been qualified in this aircraft and therefore the use of the ejection seat during the APE 1.2 evaluation will be at the discretion of the aircraft Commander. The interdepartmental communication from Mr. D.R. Segner, subject: AH-56 Ejection Seat Qualification, dated 20 May 1970, contains the controlling guidelines for the use of the ejection seat.

6. Emergency Procedures:

a. Checklist Emergency Procedures: The emergency procedures detailed in POMM 55-1520-22-10 CL, (January 1971), Operator's and Crewmember's Checklist, for aircraft serial no. 66-8834 shall be followed with special emphasis on the following:

(1) Prop System Control Failure - page N9

(2) Proximity Device Warning - page E20. This system has been removed.

(3) Stick Centering Malfunction/Failure - pages E26, and E27.

(4) N_f Control Failure - page E28.

b. Additional Emergency Procedures. The following emergency procedures not included in the pilot's checklist should be followed:

(1) In-flight emergency egress from the cockpit should be out the righthand side to avoid possible contact with the tail rotor.

(2) Low transmission oil pressure (below 80 psi) does not require an emergency landing. Flight may be continued for a maximum of 30 min. at 90 - 100 KIAS if the following transmission oil pressure vs. transmission oil temperature conditions are not exceeded:

80 psi @ 113°C

78 psi @ 120°C

75 psi @ 130°C*

72 psi @ 140°C

* Flight above 130°C is limited to 10 minutes.

(3) Engine oil overtemperature due to oil cooler fan failure does not require an emergency landing if engine oil pressure is above 40 psi and an airspeed of 130 KIAS can be maintained.

(4) Emergency Landing Gear Extension. Emergency extension of the main landing gear is accomplished by an accumulator system designed to function at 100 knots calibrated airspeed or less.

c. Emergency Hover. This flight condition should be limited to the absolute minimum time necessary.

7. Cautions and Warnings:

a. Caution. Blade moment stall has been encountered at conditions of 7° indicated collective pitch blade angle, 120 knots calibrated airspeed, and load factors of 1.8 or greater. This condition is characterized by right roll and pitch up. Recovery techniques shall be consistent with the procedures demonstrated to USAASTA pilots by Lockheed during the pilot training.

b. Caution. During Pre-Engine Start System Checks insure that the RPM Set Switch (N_f Beeper) has been set in the DECR position for a minimum of five seconds.

c. Caution. Do not apply rotor brake with engine running. Apply rotor brake only below 40% N_R with engine off and TIT below 320°C. Rotor brake may be applied before engine start but must be released at powers greater than ground idle. Do not attempt to keep rotor brake on beyond ground idle when running up.

d. Warning. Do not start APU with rotor running and a known or suspected No. 1 hydraulic system malfunction at any time.

e. Warning. Landing roll deceleration must be accomplished using reverse propeller thrust and main gear braking only. Aft cyclic inputs during ground operation can overstress main rotor control components or airframe structure.

f. Warning. Avoid operation at 40°F or below with visible moisture present.

8. Limited Life Parts:

a. The maximum allowable operating times (MAOT) for fatigue critical component parts are as listed in Table 1, Incl 1.

b. USAASTA personnel shall assure that the special inspection indicated under the S.I. column of Table are performed at the intervals specified in document IDC 87-71-045, "Tracking of Finite Life and TBO Items", dated 20 Jan 1971.

9. Propeller Blade Angle Limitations:

a. Maximum allowable propeller blade angle during climb below 100 knots calibrated airspeed is +30°.

b. Maximum propeller blade angle is +40.8 degrees.

c. Minimum propeller blade angle is -12° (with counterweights).

10. Safe Take-Off Corridor: Note: Contractor has demonstrated throttle chops from hover to 60 knots at 7 to 10 feet wheel heights.

11. Preliminary Operator's Manuals. The helicopter shall be operated in accordance with the Preliminary Operator's Manual POMM 55-1520-222-10 dated 15 March 1969 except that Chapter 7 shall be disregarded and the operating limitations set forth in this flight release shall apply. The pilots checklist POMM 55-1520-222-10CL, dated January 1971, with the two additions listed below, shall be used:

a. Tail wheel lock pin check.

b. Fuel control shaft position check to insure that it is at the shutoff step.

12. Pilot Station. The pilot station shall be the forward cockpit.

13. Flight in Turbulence. Flying is restricted to less than moderate turbulence, that is, approximately 90% of airspeed fluctuation occurrences due to turbulence not to exceed ±5 knots and approximately 10% of such airspeed fluctuations not to exceed 8 knots and/or approximately 90% of c.g. vertical load factor excursion occurrences due to turbulence not to exceed ±0.3g and approximately 10% of such load factor excursions not to exceed ±0.5g.

FOR THE COMMANDER:

s/Robert F. Forsyth LTC
CHARLES C. CRAWFORD, JR.
Director of Flight Standards
and Qualification

7 Incl
as

TABLE I

ITEM	PART NUMBER	COMPONENT	MAOT	S.I.
1.	1019765-301	M/R Blade Assy	425	15.4
				180.4
2.	1018578-303	M/R Movable Hub Assy	400	30.9
				60.2
				180.11
3.	2602853	M/R Tension Torsion Pak	200	
4.	LS10066-1	M/R T-T Pak Pin (8)	1920	15.8
				180.2
5.	LS10055-3	M/R Hinge Bolt	450	30.4
6.	740290-107	M/R Hinge Bearing	60	30.4
7.	1019772-301	M/R Fixed Hub Assy	480	30.8
8.	1010460-305	M/R Gyro Dome	440	60.7
9.	1013452-301	M/R Gyro Drive Assy	680	
10.	1013447-301	M/R Gyro Hub	250	
11.	1019896-101	M/R Gyro Arms	2700	
12.	1021500-301	M/R Pitch Arm	300	2.3
				30.7
				50.2
13.	1010335-325	M/R Servo Package	380	
14.	1019881-105	M/R Coll Bellcrank	260	
15.	1019989-101	M/R Coll Link	200	
16.	1013450-307	Rotating Push Rod	900	
17.	1009430-301	Rotating Swashplate Assy	900	
18.	1011572-305	Non-Rotating Swashplate	2600	
19.	1013916-101	Swashplate Ball Spline	1350	
20.	1001981-107	Ball Spline Housing	900	
21.	1013432-305	M/R Pitch Link	1570	60.1
22.	100723A3	M/R Collective Dampers	450	
23.	673107-101	M/R Cyclic Dampers	385	
24.	1000156-177	Main Transmission (Sprag)	150*	2.1
				30.6
25.	1000211-103	M/R Mast (Trans. 1010)	1340	
26.	1020252-105	Sprag Clutch	50	25.1
27.	670869-103	A.P.U.	1800	
28.	670825-103	Lube Pump	150	
29.	671766-301	A.P.U. Clutch	1200	
30.	740530-105	A.P.U. Drive Shaft	1200	
31.	T-64-GE-16	Engine	150	50.1
32.	1020251-111	Engine Torqueshaft	50	25.2
33.	671221-301	Nash Actuator Fuel Control	1200	
34.	1020263-101	Engine Speed Control	50	25.1
35.	1002063-101	Propeller Drive Shaft	1200	
36.	738700-1	Propeller Gearbox	400	

* Exclusive of sprag clutch.

ITEM	PART NUMBER	COMPONENT	MAOT	S.I.
37.	11FA1CB8-0	Propeller Blades	1200	
38.	738620	Propeller Actuator	1200	
39.	1015146-301	T/R Drive Shaft Assy	265	
40.	1007925-303	T/R Gimbal Ring Assy	200	
41.	1008840-303	T/R Spindle Mechanism	300	
42.	1009256-301	T/R Hub Mech. Assy	1310	
43.	1019382	T/R Spider - Pitch Control	300	
44.	1019370-301	T/R Blade Assy	560	
45.	1009327-301	T/R Feather Arm	1200	
46.	1008898-305	T/R Spline Shaft	280	Inwork
47.	1001707-301	M/R Blade Attach Bolt,Fwd	470	180.1
48.	1018922-103	M/R Blade Attach Bolt,Aft	470	
49.	1020319-101	M/R Collective Clevis	2380	
50.	1014921-301	M/R Cyclic Clevis	2150	
51.	1014055-309	Pushrod Assy-Non Rotating	2970	
52.	1014055-310	Pushrod Assy-Non Rotating	2970	
53.	1013915-303	Pushrod Assy-Non Rotating	2970	
54.	1020312-101	Cone	2600	
55.	1009989-301	Retainer	2970	
56.	1020183-101	Link Assy	2600	
57.	1015595-313	Link Assy	2600	
58.	740596-301	T/R Tension Torsion Pack	1310	
59.	1010445-301	T/R T.T Pack Pin	1310	
60.	1009338-301	T/R T T. Pack Pin	1310	
61.	1007712-301	T/R Spindle Support Housing	200	
62.	1008361-311	T/R Drive Spindle	110	
63.	740599-301	T/R Pitch Link	900	
64.	1008899-301	T/R Bell Crank	340	
65.	1007460-301	Bellcrank, Direct Control	2070	
66.	1007462-301	Support, Direct Control	2070	
67.	1007464-301	Link Assy,Direct Control	2070	
68.	1009046-101	Rod Assy, Direct Control	2070	
69.	1001690-115	Directional Servo Assy	1200	
70.	1000663-101	Fitting, Front Beam	320	60.8 60.10
71.	1000665-101	Fitting, Rear Beam	320	60.8 60.10
72.	740882-301	T/R Feather Bearing		5.1**
73.	740883-301	Tail Rotor Feather Bearing		5.1**
74.	740597-303/305	T/R Drive Spindle Bearing	30/60	2.5**,****
75.	740718-301/303	T/R Gimbal Bearing	30/60	2.5**,****
76.	670925-105	Oil Cooler Fan-Vickers		**,***

** Special inspection and replacement procedure based on cumulative wear.
*** Visual inspection for leakage prior to and after each flight.
**** Daily Inspection.

ITEM	PART NUMBER	COMPONENT	MAOT	S.I.
77.	1015276-301	Spider Assy - U-Joint	680	120.5
				30.17
78.	1002064-103	Prop Drive Shaft Coupling	6300	1.13
79.	1002072-101	Prop Drive Shaft Coupling	6300	1.13
80.	671083	Cooling Fans	1800	
81.	671084	Cooling Fans	1800	
82.	671085	Cooling Fans	1800	

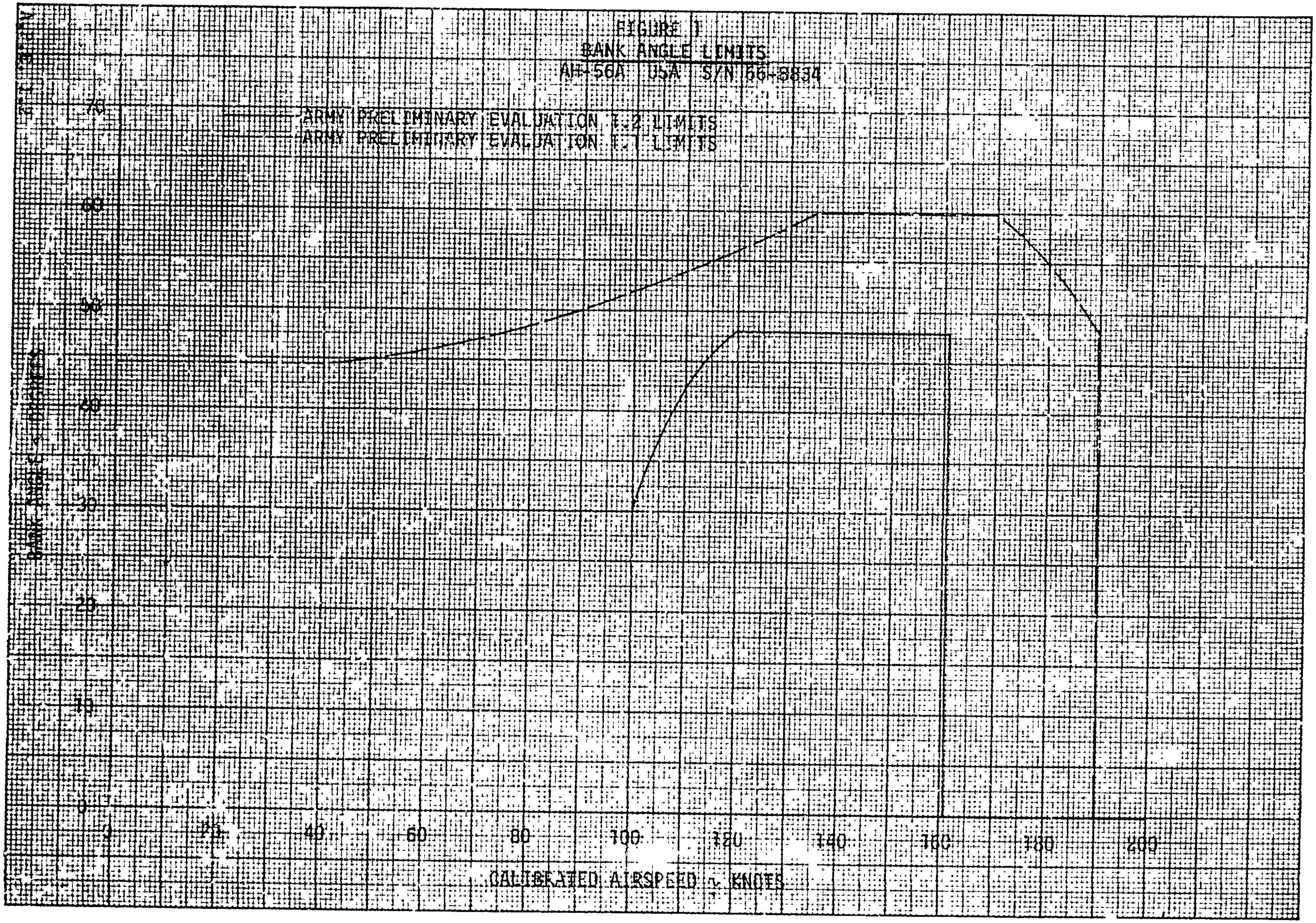


FIGURE 1
BANK ANGLE LIMITS
AH-56A USA S/N 66-8834

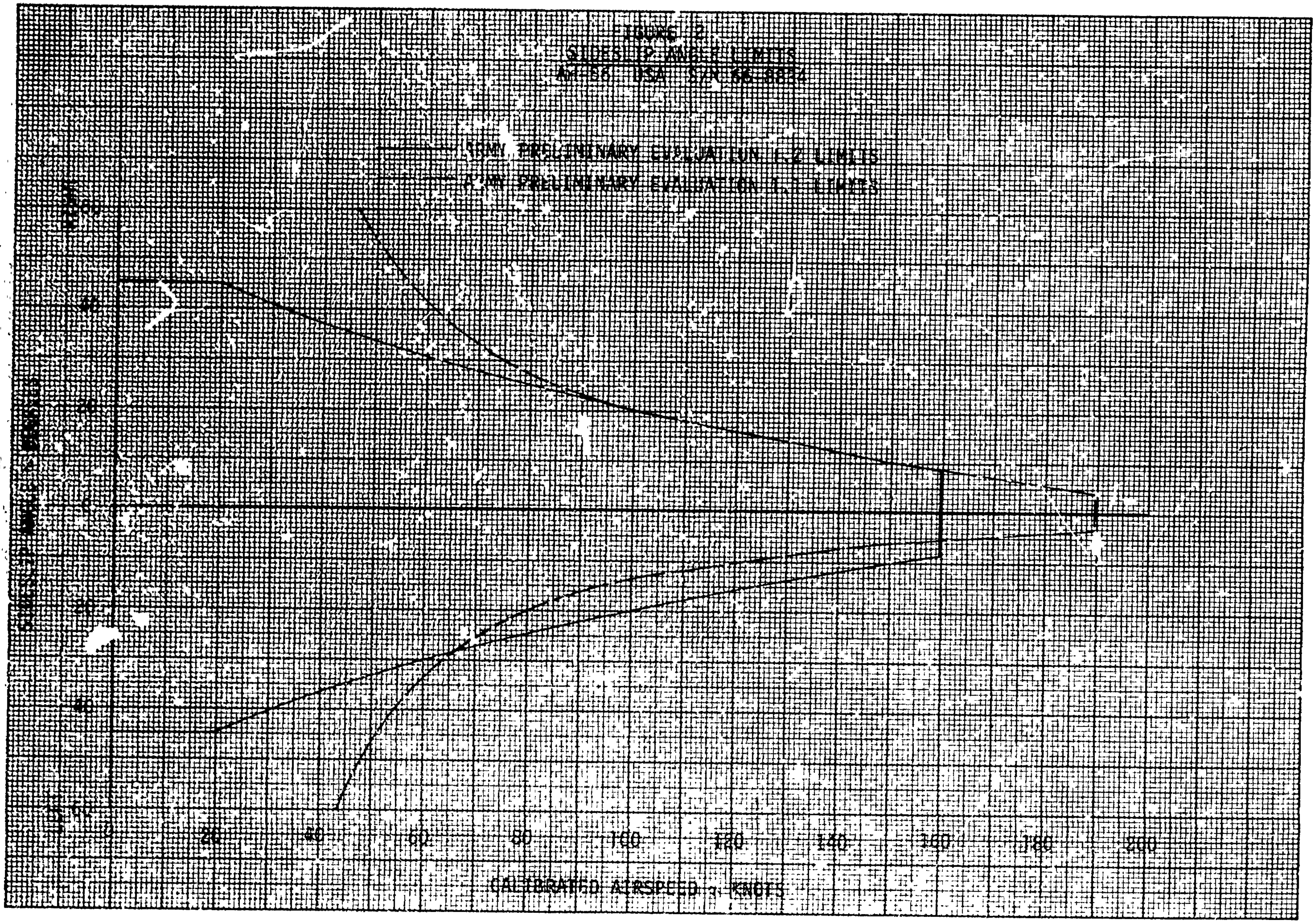


FIGURE 2
SIDESLIP ANGLE LIMITS
AH-56 USA S/N 66-8834
ARMY PRELIMINARY EVALUATION 1.2 LIMITS
ARMY PRELIMINARY EVALUATION 1.1 LIMITS
60
40
20
0
20
40
60
0
20
40
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 3
SUSTAINED LOAD FACTOR LIMITS
AH-56A USA S/N 66-8834

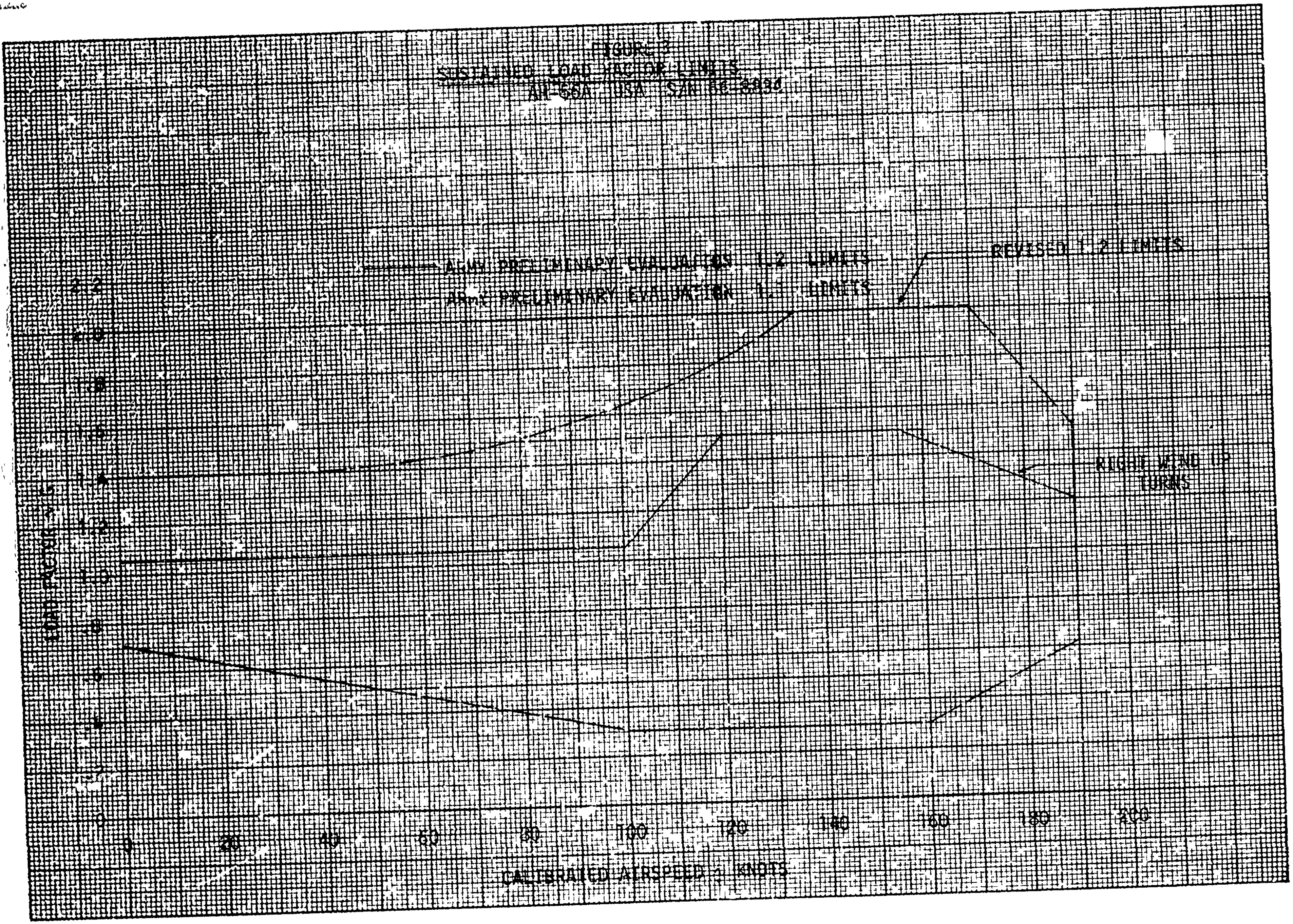


FIGURE [illegible]
COLLECTIVE BLADE ANGLE LIMITS
AH-56A USA S/N 66-8834

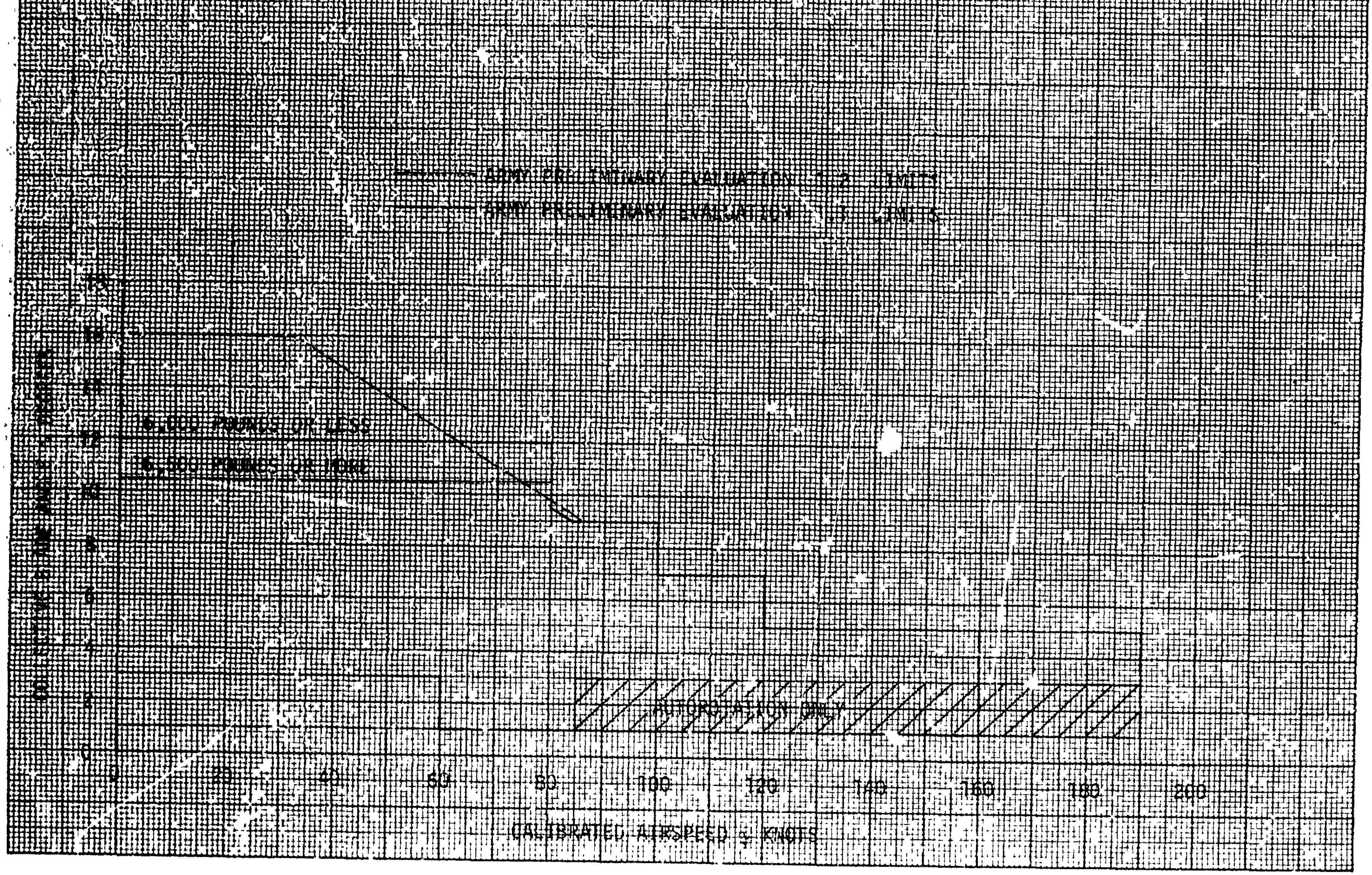
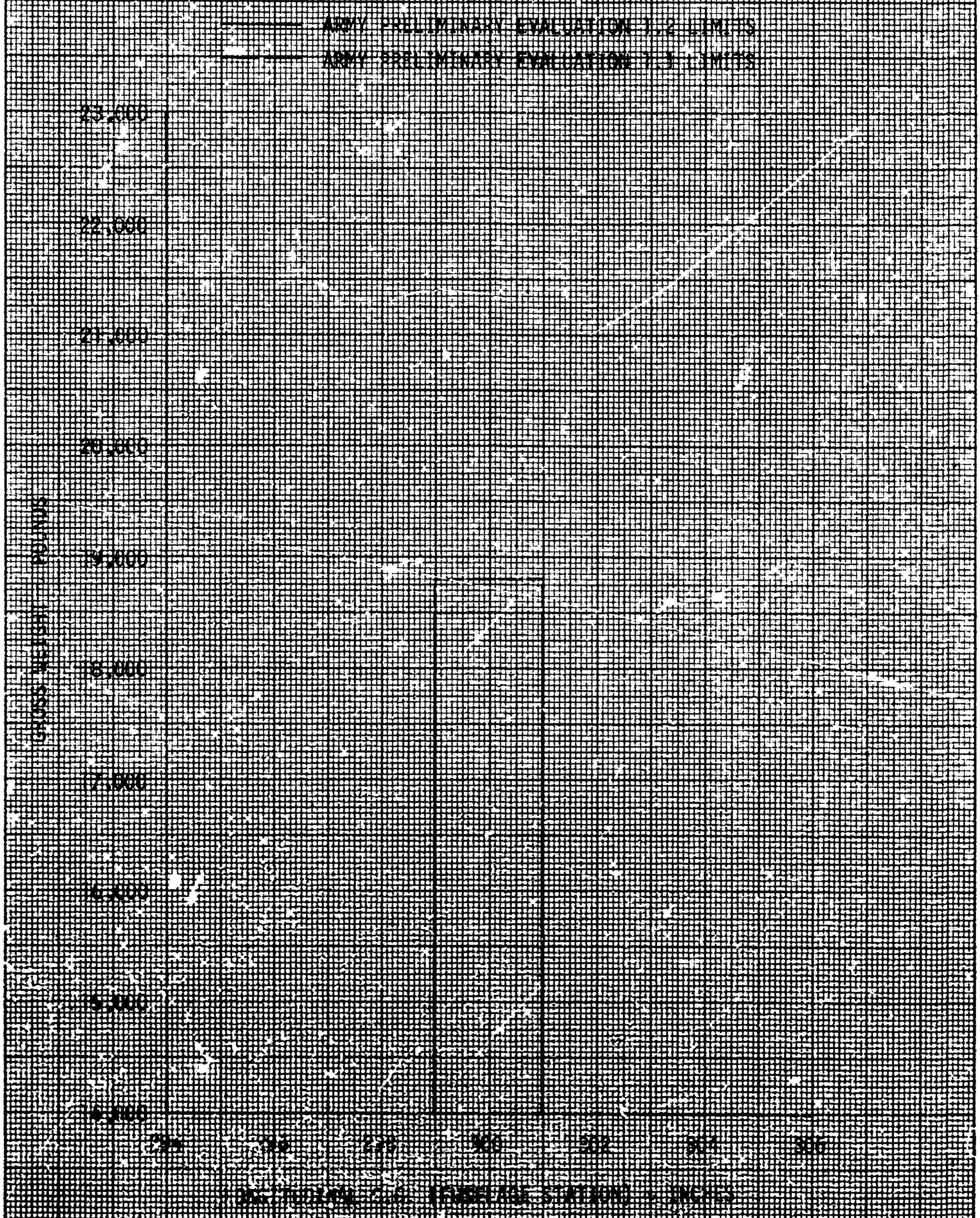


FIGURE 5
LOADING LIMITS
AH-56A USA S/N 66-8834



NNNNPTTUZYUW RUWTFFAØ923 1112Ø52-UUUU-RUWJBDA

ZNR UUUUU

P R 2116ØØZ APR 71

FM CG USAAVSCOM STL MO

TO RUWJBDA/CO USAASTA EDWARDS AFB CALIF

INFO RUEBBNA/CG USAMC

RUWJREA/CO LOCKHEED PLANT ACTIVITY VAN NUYS CALIF

BT

UNCLAS
AMSAV-R-F

ACTION FOR SAVTE-P INFO AMC FOR AMCRD-FQ AND AMCSF-A

SUBJECT: AH-56A APE 1.2 SAFETY OF FLIGHT RELEASE

IN REPLY REFER TO: Ø4-26

A. TELECON BETWEEN MR. D. MACPHERSON, ASTA AND LTC FORSYTH, 2Ø APR 71, CONCERNING THE EXTERNAL STORES JETTISON CAPABILITY OF AIRCRAFT 1ØØ9 DURING APE 1.2 TESTING.

B. LETTER AMSAV-R-F, DATED 19 APR 71, SUBJECT: APE 1.2 SAFETY OF FLIGHT RELEASE.

1. PER YOUR REQUEST, PARA 2D AND 2E OF REFERENCE B ARE HEREBY RESCINDED. IT IS UNDERSTOOD THAT ALL APE 1.2 TESTING WILL BE CONDUECTED WITH THE EXTERNAL STORES MOUNTING RACKS SECURED IN A NON-JEETISONABLE CONDITION. 2. CHANGE PARA 6B(2) OF REFERENCE B TO READ:

PAGE 2 RUWTFFAØ923 UNCLAS

LOW TRANSMISSION OIL PRESSURE (BELOW 8Ø PSI) DOES NOT REQUIRE AN EMERGENCY LANDING. FLIGHT MAY BE CONTINUED FOR A MAXIMUM OF 3Ø MINUTES AT 9Ø - 1ØØ KIAS USING THE FOLLOWING LIMITS:

77 PSI ---- 85 DEGREES -- 13Ø DEGREES C"

73 PSI ---- 9Ø DEGREES -- 13Ø DEGREES C"

7Ø PSI ---- 75 DEGREES -- 13Ø DEGREES C"

FLIGHT ABOVE 13Ø DEGREES C IS LIMITED TO 1Ø MINUTES.

3. CHANGES TO INCL 1, PAGE 1.

ITEM	P/N	COMPONENT	MAOT	S.I.
26	1Ø2Ø252-1Ø5	SPRAG CLUTCH	15Ø	25.1
32	1Ø2Ø251-111	ENGINE TORQUESHAFT	15Ø	25.2 (5ØHR)
34	1Ø2Ø263-1Ø1	ENGINE SPEED CONTROL	15Ø	25.1 (5Ø HR)

4. ADD THE FOLLOWING TO FOOTNOTES ON PAGE 2, INCL 1.

'''''REMOVE AND REPLACE AT 5Ø HR INSPECTIRN INTERVAL.

5. ADD PAGE 3, INCL 1.

ITEM	P/N	COMPONENT	MAOT	S.I.
83	741357	SPRAG CLUTCH BEARINGS	5Ø	'''''

BT

Ø923

NNNNRTTUZYUW RUWTFFA 1668 1191849-UUUU-RUWJBDA.

ZNR UUUUU

R 291835Z APR 71

FM CG USAAVSCOM ST LOUIS MO

TO RUWJBDA/CO USAASTA EDWARDS AFB CALIF

INFO RUWJREA/CO US ARMY LOCKHEED PLANT ACTIVITY VAN NUYS CALIF

RUWJHUA/CHIEF AAWS PROJ MGR FIELD OFC YPG, YUMA ARIZONA

RUEBBNA/CG USAMC

BT

UNCLAS

AMSAV-R-F

ACTION FOR SAVTE-P INFO FOR LOCKHEED ACT, ATTN SAVLO YUMA FIELD OFC, ATTN LTC VAUGHN AMC FOR AMCSF-A AND AMCRD-FQ

SUBJECT AH-56A HELICOPTER APE 1.2

IN REPLY REFER TO AMSAV-R-FT Ø4-36

A. ASTA TWX SAVTE-T ØØØ37, AH-56 HELICOPTER APE 1.2, 2Ø162ØZ APRIL 1971.

B. AVSCOM TWX AMSAV-R-FT Ø4-32, AH-56A HELICOPTER APE 1.2 SCOPE OF TEST, DATED 2317ØØZ APRIL 1971.

1. REFERENCE ASTA TWX PROPOSED SCOPE OF TEST FOR APE 1.2 WHICH WAS APPROVED BY AVSCOM TWX WITH EXCEPTION OF AUTOROTATIONAL ENTRIES. IN VIEW OF MAINTENANCE PROBLEMS AND APE II THE SCOPE IS CHANGED AS

FOLLOWS

A. AIRCRAFT WILL BE RETURNED TO LCC FOR MANEUVER/STABILITY (RIGHT WIND-UP TURNS) TO 2.ØG, STORES ON TO INVESTIGATE BLADE MOMENT STALL (BMS). IF BMS NOT ENCOUNTERED WITH STORES ON THEN INVESTIGATE WITH STORES OFF. AVSCOM -- YUMA REP AND ASTA TO WITNESS LCC TESTS AND DATA.

B. AFTER LCC TESTS ASTA WILL FLY MANEUVER/STABILITY TO G LIMIT AND AIRSPEED ESTABLISHED BY ON-SITE AVSCOM - YUMA REP.

C. LONG. TRIM CHARACTERISTICS FLIGHTS MAY BE DELETED.

2. UPON COMPLETION OF FLIGHTS IN 1.B, APE 1.2 IS COMPLETED.

BT

1668

NNNNRTTUZYUW RUWTFFA 1669 1191849-UUUU--RUWJBDA.

ZNR UUUUU

R 291930Z APR 71

FM CG USAAVSCOM ST LOUIS MO

TO RUWJBDA/CO USAASTA EDWARDS AFB CALIF

INFO RUWJREA/CO US ARMY LOCKHEED PLANT ACTIVITY VAN NUYS CALIF

RUWJHUA/CHIEF AAWS PROJ MGR FIELD OFC YPG, YUMA ARIZONA

RUEBBNA/CG USAMC

BT

UNCLAS

AMSAV-R-F

ACTION FOR SAVTE-P INFO FOR LOCKHEED ACT, ATTN SAVLO YUMA FIELD OFC,

ATTN LTC VAUGHN AMC FOR AMCSF-A AND AMCRD-FQ

SUBJECT AH-56A HELICOPTER APE 1.2

IN REPLY REFER TO AMSAV-R-F 04-37

A. AVSCOM TWX AMSAV-R-FT 04-36, AH-56A HELICOPTER APE 1.2, DATED 291835Z APR 71.

IN ACCORDANCE WITH THE CONTENTS OF REFERENCE A THE MANEUVERING G LIMITS, BASED ON THE RESULTS OF LCC TESTS AND DATA, ARE ESTABLISHED AT 20.G FOR BOTH LEFT AND RIGHT WIND UP TURNS.

BT

1669

NNNNPTTUZYUW RUWTFFA6444 2712Ø54-UUUU--RUWJBDA.

Z NR UUUUU

P 28192ØZ SEP 71

FM CG USAAVSCOM STL MO

TO RUWJBDA/CO USAASTA EDWARDS AFB CALIF

INFO RUWJREA/CO US ARMY LOCKHEED PLANT ACT VAN NUYS CALIF

RUWJHUA/CHIEF AAWS PROJ MGR YUMA FIELD OFC YPG YUMA ARIZ

BT

UNCLAS

AMSAV-EF

ACTION FOR: SAVTE-P, INFO FOR LOCKHEED, CO: YUMA FIELD OFC, ATTN: ASTA TEST TEAM AND AAWS PM REP AT YUMA.

SUBJECT: AH-56A SAFETY OF FLIGHT RELEASE FOR RDAT I

IN REPLY REFER TO: AMSAV-EF

A. REFERENCE AVSCOM LETTER AMSAV-EF, SUBJECT: APE 1,3 SAFETY OF FLIGHT RELEASE, DATED 1 SEPT 71.

THIS TWX CONSTITUTES A SAFETY OF FLIGHT RELEASE FOR THE ASTA PORTION OF RDAT I FOR THE AH-56A TEST PROGRAM. THIS FLIGHT RELEASE IS CONTINGENT UPON THE USE OF THE FLIGHT ENVELOPE, LIMITATIONS, CAUTIONS AND WARNINGS AS INCLUDED IN THE REF LETTER FOR THIS TEST PROGRAM. EXCEPTIONS TO THE LETTER ARE AS FOLLOWS:

A. CHANGE A/C SERIAL NUMBER TO 66-8831.

B. DELETE THE TELEMETRY REQUIREMENTS BETWEEN A/C AND TM VAN.

C. DELETE REFERENCE TO THE EJECTION SEAT.

D. -1ØCL PUBLICATION DATE IIS CHANGED FROM JAN 71 TO JULY 71.

E. DURING WEAPONS FIRING THE PROCEDURES AND SAFETY PRECAUTIONS SET FORTH IN THE INTERIM SAFETY STATEMENT FOR RDAT I DATED 28 SEPT 71 APPLY.

BT

6444

NNNNPTTUZYUW RUWTFFA 768Ø 2812135-UUUU--RUWJBDA.

ZNR UUUUU

P Ø821ØØZ OCT 71

FM CG USAAVSCOM ST LOUIS MO

TO RUWJBDA/CO USAASTA EDWARDS AFB CALIF

INFO RUWJREA/CO US ARMY LOCKHEED PLANT ACT VAN NUYS CALIF

RUWJHUA/CHIEF, AAWS PROJ MGR YUMA FIELD OFC, YUMA ARIZ

RUEBBNA/CG USAMC

BT

UNCLAS

AMSAV-EF

ACTION FOR: SAVTE-P, INFO FOR: LOCKHEED, CO: YUMA FIELD OFC, ATTN: ASTA TEST TEAM AND AAWS PM REP AT YUMA, AMC FOR AMCRD-FQ AMCSF-A

SUBJECT: APE 1.3 SAFETY OF FLIGHT RELEASE

IN REPLY REFER TO: AMSAV-EFT 1Ø-11

A. REFERENCE LETTER FROM AMSAV-EF TO SAVTE-P, SUBJECT: APE 1,3 SAFETY OF FLIGHT RELEASE. 1 SEPT 71.

PARAGRAPH 3.E OF REFERENCED SAFETY OF FLIGHT RELEASE IS REVISED TO INDICATE THAT MINIMUM PROPELLER BETA ANGLE IS -5 DEGREES (INSTEAD OF PLUS 8 DEGREES) FOR DIVES.

BT

768Ø

DEPARTMENT OF THE ARMY
US ARMY AVIATION SYSTEMS COMMAND
PO Box 209, St. Louis, Missouri 63166

AMSAV-EF 1 Sep 1971

SUBJECT: APE 1.3 Safety of Flight Release

Commanding Officer
U.S. Army Aviation
Systems Test Activity
ATTN: SAVTE-P

1. This letter constitutes a safety of flight release for day V.F.R. flight of AH-56A S/N 66-8834 for the conduct of Army Preliminary Evaluation 1.3.

2. This flight release is contingent upon the following:

a. The airworthiness of all onboard flight test equipment and instrumentation being assured by a safety inspection performed by USAASTA personnel.

b. The flight control systems being rigged in accordance with drawings and specifications.

c. A functioning radio link directly between the Telemetry van and the test aircraft.

d. Proper functioning of flight control augmentation equipment, specifically the pitch desensitizer and roll compensator units.

3. The authorized flight envelope is as described below.

a. Airspeed Limitations.

(1) Forward Flight . . . The maximum authorized forward flight speed is shown in Figures 1 and 2, Incl 1.

(2) Landing Gear Extended . . . The maximum authorized flight speed for normal landing gear extension (or with the landing gear extended) is 130 knots calibrated airspeed.

(3) Butterfly Canopy (Forward and/or Aft) . . . The canopy open (forward and/or aft) condition is authorized only for ground conditions, rotor stationary, and winds of 45 knots or less.

(4) Taxi, Takeoff, and Landing. . .

(a) Tail Wheel Unlocked. . . The maximum authorized taxi speed with the tail wheel unlocked is 20 knots.

(b) Tail Wheel Locked. . . The maximum authorized taxi speed with the tail wheel locked is 70 knots calibrated airspeed.

(5) Sideward and Rearward Flight. . .

(a) Gross weight, 18,300. . . 35 knots sideward, 30 knots rearward.

(b) Gross weight, 20,500. . . 16 knots sideward and 16 knots rearward.

(6) Run-on Landings. . . The authorized maximum airspeed at touchdown is limited to 70 knots calibrated airspeed.

(7) Autorotative Descent. . . Stabilized autorotative descent airspeed shall be limited to 85 to 95 knots calibrated airspeed.

b. Collective Blade Angle. . . Collective/main rotor swashplate position is sensed and presented on a cockpit display in degrees. The authorized cockpit displayed collective angles as a function of airspeed are shown in Figure 3, Incl 2.

c. Bank Angle Limitations. . .

(1) The maximum authorized transient bank angle is 70°, with load factor not exceeding that shown in Figures 1 and 2, Incl 1, for a discrete airspeed.

(2) The maximum authorized sustained bank angle as a function of airspeed will be commensurate with that permitted by the Load Factor Airspeed Envelopes shown in Figures 1 and 2, Incl 1.

d. Sideslip Envelope. . . The maximum authorized sideslip as a function of calibrated airspeed is shown in Figure 4, Incl 3.

e. Descents . . . The maximum authorized rate of descent is 6000 feet per minute. Flight path (dive) angle is limited to a maximum of 20 degrees with a minimum propeller beta angle of +8 degrees except during landing.

f. Practice/Intentional Autorotation. . . Autorotational landings are prohibited. All intentional autorotational descents will be terminated by powered flight at a safe altitude but in no case below 500 ft. AGL.

g. Control Input Limits, Directional. . . Abrupt pedal inputs in forward flight shall not exceed ±1 inch from trim or result in sideslip angles greater than that authorized by the sideslip-airspeed envelope shown in Figure 4, Incl 3.

h. Control Input Limits, Cyclic. . .

(1) 100% N_R . . . Cyclic control inputs shall be limited to ±2 inches during ground operations.

(2) Cyclic stirs. . . Successive cyclic stirs at rates greater than one cycle in two seconds (0.5Hz) are to be avoided, 2 cps stirs of 1 (one) cycle duration are permitted.

i. Load Factor. . . The authorized load factor airspeed envelope is shown in Figures 1 and 2, Incl 1.

j. Altitude Limits. . . Flight above 10,500 ft. density altitude is prohibited.

k. Gross Weight and C.G. Limits. . . The authorized gross weight - C.G. envelope, is shown in Figure 5, Incl 4.

l. Rotor Speed Limits. . .

(1) Steady State Maneuvers, power on - 95% to 105% N_R, power off - 90% to 105% N_R.

(2) Transient Maneuvers, power on - 95% to 105% N_R, power off - 85% to 110% N_R.

m. Rotor Start/Stop Limits. . . The rotor shall not be started or stopped in winds in excess of 20 knots.

n. Touch Down Sink Rates. . . Touch down sink rate shall not exceed 9.5 feet per second at 18,300 pounds. . . (570 FPM) and 9.0 feet per second at 20,500 pounds (540 FPM)

o. Wind Limits. . . Flight operations shall not be conducted in winds in excess of 20 knots.

4. The engine, transmission, hydraulic system, and APU limitations and associated instrument markings are in accordance with the POMM 55-1520-22-10, Chapter 7, except as detailed below:

LIMITATIONS	INSTRUMENT MARKING
Gas Generator RPM	
58% idle (minimum)	red line
72% idle (maximum)	(no mark)
63 to 100% normal (run) operating range	green band
100% maximum continuous	(no mark)
100 to 101.5% time limited to 10 seconds	yellow band
101.5% inspection limitation	red line
Power Turbine RPM	
95% minimum (power on)	red line
95% to 105% normal operating range	green band
105% maximum (power on)	red line
Overspeed cutoff - 113%	(no mark)
Main Rotor RPM	
90% minimum power off (95% minimum power on)	red line
95 to 105% normal operating range	green band
105% maximum power on/off	red line

AMSAV-EF 1 SEP 1971

SUBJECT. APE 1.3 Safety of Flight Release

LIMITATIONS	INSTRUMENT MARKING
Engine Oil Temperature	
0°C minimum (power on)	red line
0 to 107°C normal operating range	green band
107°C maximum continuous	red line
107°C to150°C for 30 minutes - emergency only above 150°C (see para 6, emergency procedures)	(no mark)
Transmission Oil Temperature	
-30°C minimum	red line
0 to 113°C normal operating range	green band
113°C maximum continuous	red line
113°C to 130°C for 30 minutes and 130°C to 135°C for 10 minutes. (For emergency only and with power level equal to power for level flight at 90 to 100 KIAS)	(no mark)
Engine Oil Pressure	
10 psi minimum	red line
10 psi to 45 psi idle	(no mark)
10 psi to 75 psi normal range	green band
50 psi (minimum at 95% N_g)*	(no mark)
Transmission Oil Pressure	
70 psi minimum	red line

*When starting in cold weather, oil pressures greater than 100 psi can occur before oil temperature stabilized (within three minutes).

LIMITATIONS		INSTRUMENT MARKING
Transmission Oil Pressure (con't)		
70 to 110 psi	rmal operating range	green band
110 psi maximum		red line
Prop. Gearbox Oil Temperature		
121°C maximum		red line

5. Ejection Seat Restrictions:

The ejection seat was installed in Aircraft 66-8834 for the purpose of providing emergency egress for the contractor pilot during envelope expansion flights. The ejection seat has not been qualified in this aircraft and therefore the use of the ejection seat during the APE 1.3 evaluation will be at the discretion of the aircraft Commander. The interdepartmental communication from Mr. D. R. Segner, subject: AH-56 Ejection Seat Qualification, dated 20 May 1970, contains the controlling guidelines for the use of the ejection seat.

6. Emergency Procedures:

a. Checklist Emergency Procedures: The emergency procedures detailed in POMM 55-1520-22-10 CL, (January 1971), Operator's and Crewmember's Checklist, for aircraft serial no. 66-8834, shall be followed with special emphasis on the following:

(1) Prop System Control Failure - page E8.

(2) Stick Centering Malfunction/Failure - pages 23 and 24.

(3) Engine Control Failure - page E4 and E5.

b. Additional Emergency Procedures. The following emergency procedure not included in the pilot's checklist should be followed:

Page E26, In-flight emergency egress from the cockpit, should be out the righthand side to avoid possible contact with the tail rotor.

7. Cautions and Warnings:

a. Caution. Blade moment stall is characterized by right roll and pitch up. Recovery techniques shall be consistent with the procedures demonstrated to USAASTA pilots by Lockheed during the pilot training.

b. Caution. During Pre-Engine Start System Checks insure that the RPM Set Switch (Nf Beeper) has been set in the DECR position for a minimum of five seconds.

c. Caution. Do not apply rotor brake with engine running. Apply rotor brake only below 40% NR with engine off and TIT below 320°C. Rotor brake may be applied before engine start but must be released at powers greater than ground idle. Do not attempt to keep rotor brake on beyond ground idle when running up.

d. Warning. Do not start APU with rotor running and a known or suspected No. 1 hydraulic system malfunction at any time.

e. Warning. Landing roll deceleration must be accomplished using reversed propeller thrust and main gear braking only. Aft cyclic inputs during ground operation can overstress main rotor control components or airframe structure.

f. Warning. Avoid operation at 40°F or below with visible moisture present.

8. Limited Life Parts:

a. The maximum allowable operating times (MAOT) for fatigue critical component parts are as listed in the current AH-56A MAOT list.

b. USAASTA personnel shall assure that the special inspections indicated under the S. I. column of the MAOT list are performed at the intervals specified.

9. Propeller Blade Angle Limitations:

a. Maximum allowable propeller blade angle during climb below 100 knots calibrated airspeed is +30°.

b. Maximum propeller blade angle is +40.8 degrees.

c. Minimum propeller blade is -5°.

10. Preliminary Operator's Manuals. The helicopter shall be operated in accordance with the Preliminary Operator's Manual POMM 55-1520-222-10,

dated July 1971, except that the operating limitations set forth in this flight release shall apply where it differs from CH 7 of the operators manual. The pilots checklist POMM 55-1520-222-10 CL, (January 1971), Operator's and Crewmember's Checklist, for aircraft serial no. 66-8834, with annotated updating furnished by contractor, shall be used.

FOR THE COMMANDER:

s/Robert F. Forsyth
ROBERT F. FORSYTH
LTC, TC
Actg Chief, Flt Stds & Qual Div
Directorate for RD&E

4 Incl
as

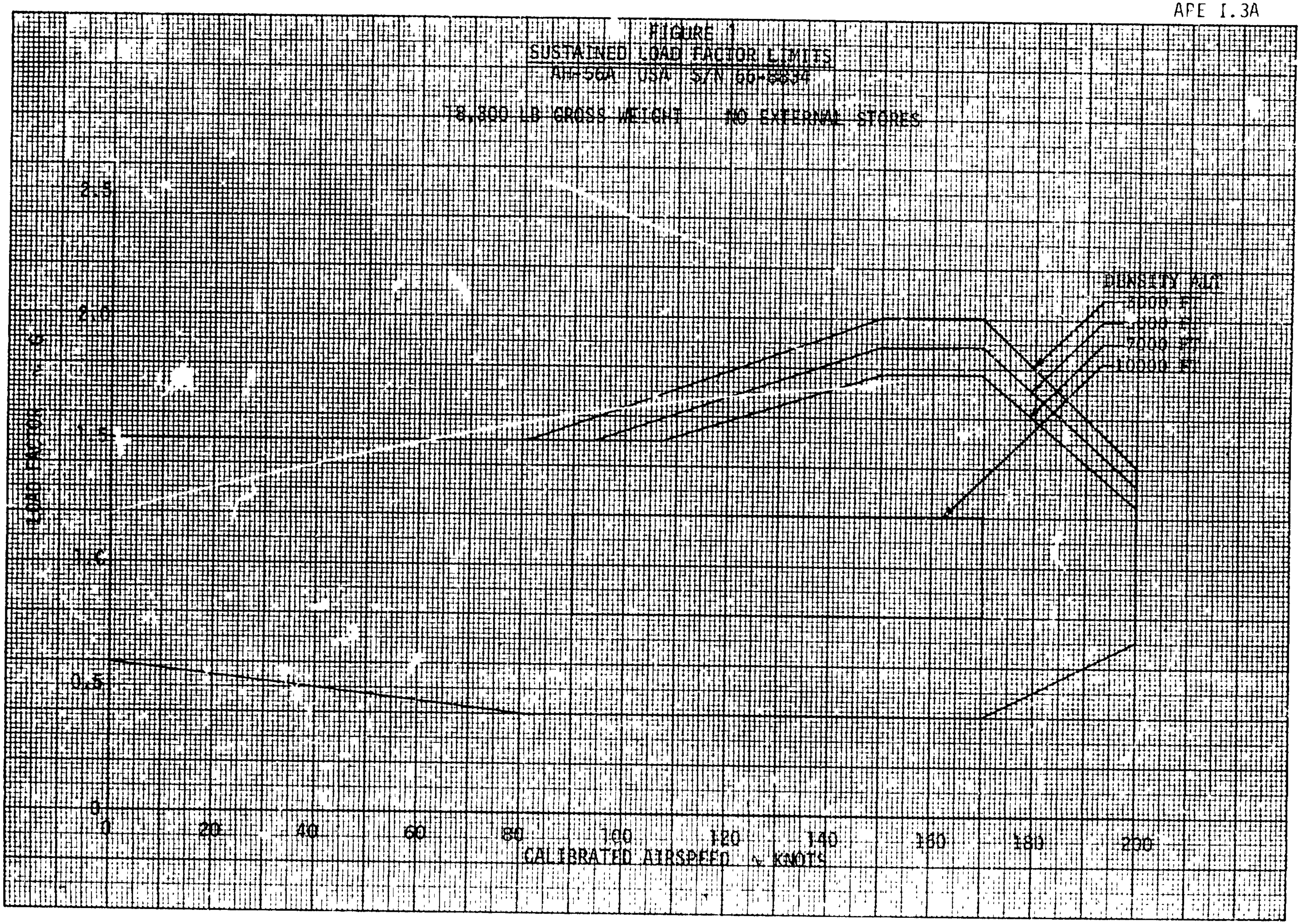


FIGURE 1
SUSTAINED LOAD FACTOR LIMITS
AH-56A USA S/N
18,300 LB GROSS WEIGHT NO EXTERNAL STORES
DENSITY ALT
7000 FT
10000 FT
LOAD FACTOR ~ G
2.5
2.0
1.5
1.0
0.5
0
20
40
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

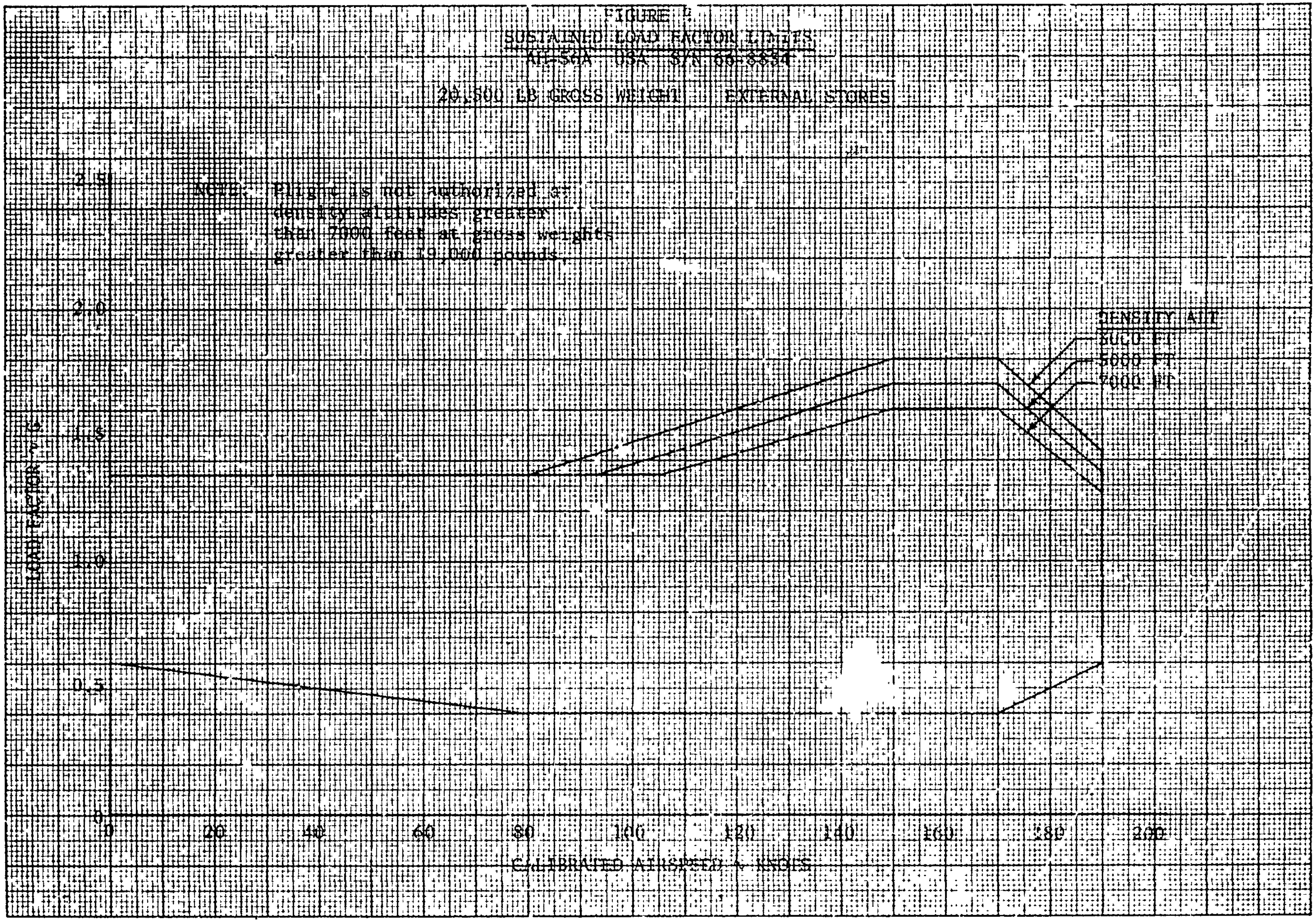


FIGURE 8
SUSTAINED LOAD FACTOR LIMITS
AH-56A USA S/N 66-8834
20,500 LB GROSS WEIGHT EXTERNAL STORES
NOTE: Flight is not authorized at density altitudes greater than 7000 feet at gross weights greater than 19,000 pounds.
DENSITY ALT
3000 FT
5000 FT
7000 FT
LOAD FACTOR ~ G
2.5
2.0
1.5
1.0
0.5
0
0
20
40
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 3
COLLECTIVE BLADE ANGLE LIMITS
AH-56A CSA S/N 66-8834

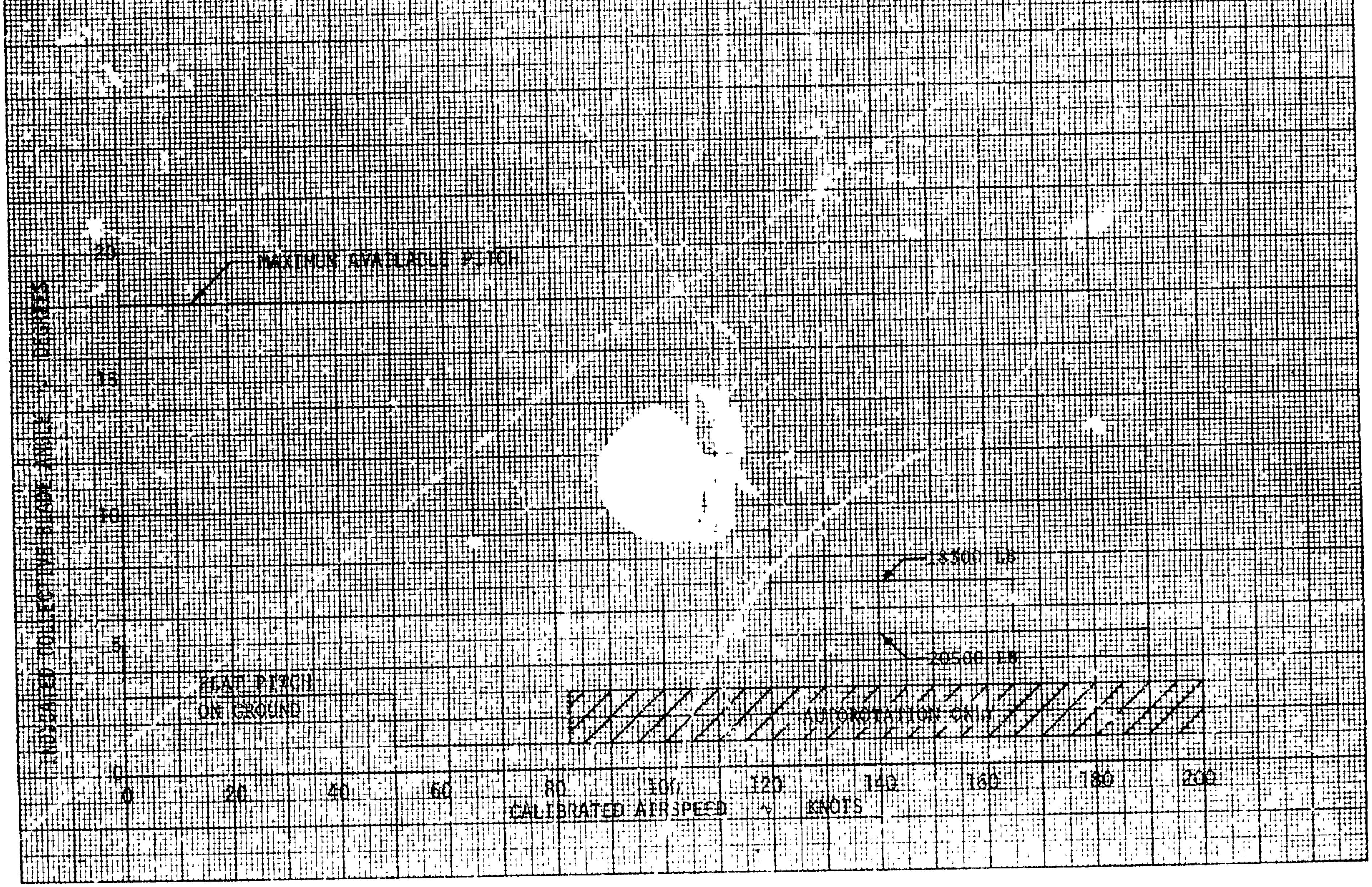
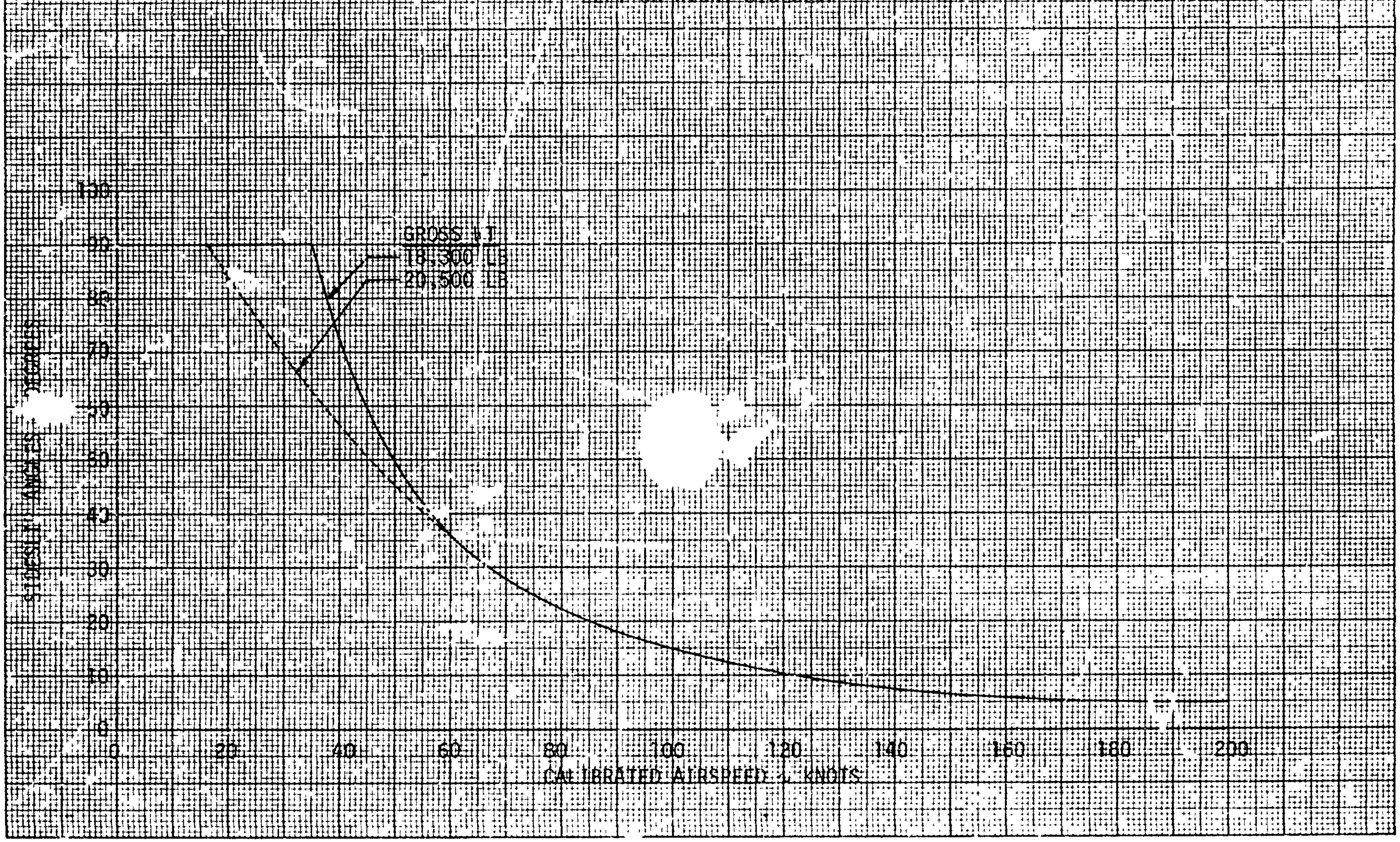


FIGURE 4
SIDESLIP ANGLE LIMITS
AH-56A USA S/N 66-8834

LEFT OR RIGHT SIDESLIP



DEPARTMENT OF THE ARMY
Headquarters, US Army Aviation Systems Command
PO Box 209, St. Louis, MO 63166

AMSAV-EF 2 DEC 1971

SUBJECT: Revision of APE 1.3 Safety of Flight Release

Commanding Officer
US Army Aviation
Systems Test Activity
ATTN: SAVTE-P
Edwards Air Force Base, Calif.

1. Aircraft modifications incorporated since the APE 1.3 Safety of Flight Release and subsequent flight test by Lockheed to substantiate said modifications and expand the previous flight envelope have resulted in the following revisions to the APE 1.3 Safety of Flight Release. Caution shall be exercised at the high airspeeds due to weak dynamic and maneuvering stability evidenced during contractor tests in this regime.

2. The following revisions shall be made to the APE 1.3 Safety of Flight Release:

a. Figures 1 through 4 have been revised and are included herein.

b. Page 2, paragraph 3.a.(5) (b) - change 16 knots to 20 knots.

c. Rotor Speed Limits (page 3, paragraph 3 L (1)), Steady State Maneuvers, power on shall be changed to 98% to 105% N_r.

d. Page 6, paragraph 7 - change title to Notes, Cautions, and Warnings.

e. Page 7, paragraph 7 e - change warning to caution.

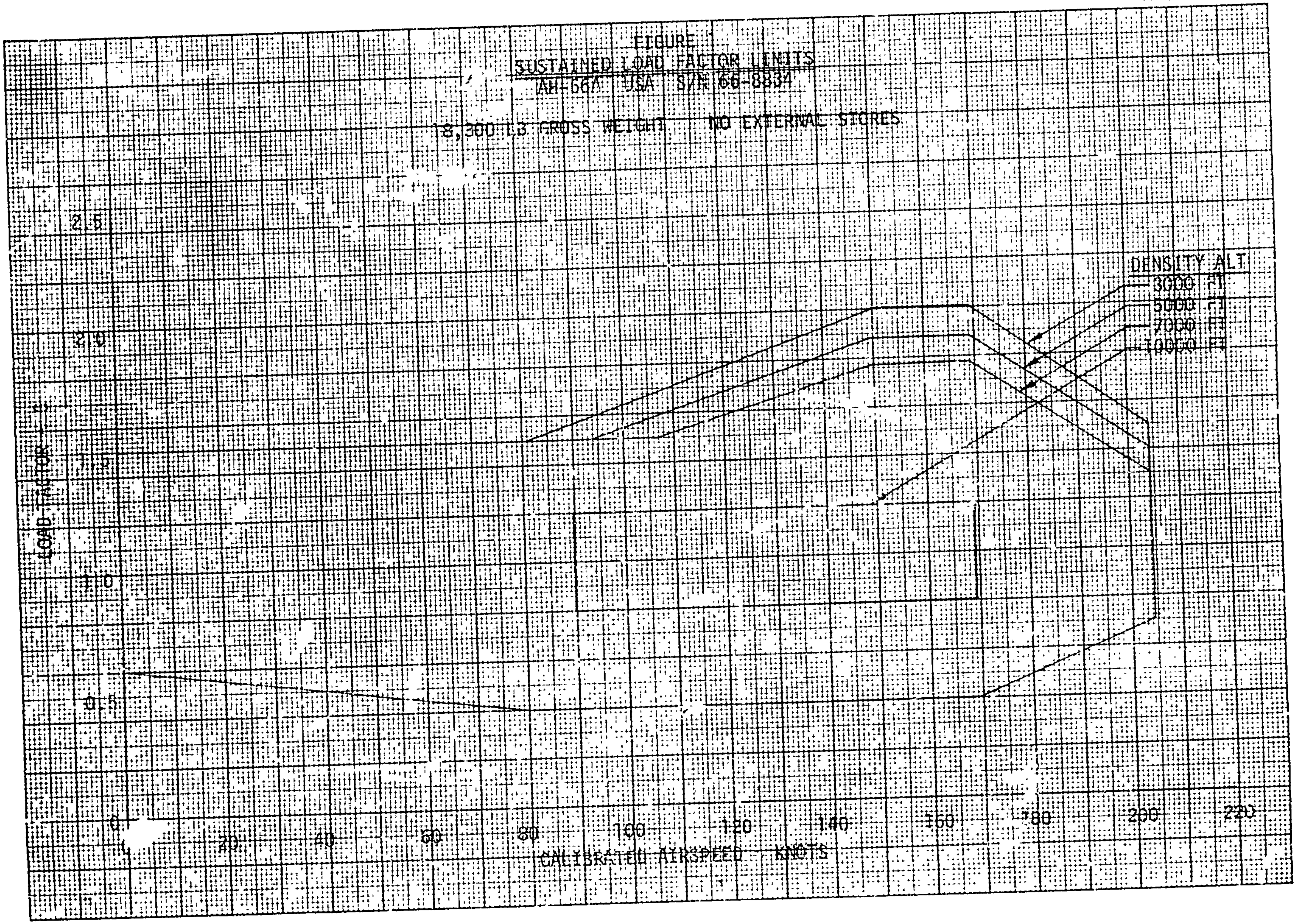
f. Page 7, paragraph 7 f - change warning to note.

FOR THE COMMANDER:

s/Robert F. Forsyth
ROBERT F. FORSYTH
LTC, TC
Asst Chief, Flt Stds & Qual Div
Directorate for Research,
Development and Engineering

FIGURE
SUSTAINED LOAD FACTOR LIMITS
AH-56A USA S/N 66-8834

18,300 LB GROSS WEIGHT NO EXTERNAL STORES



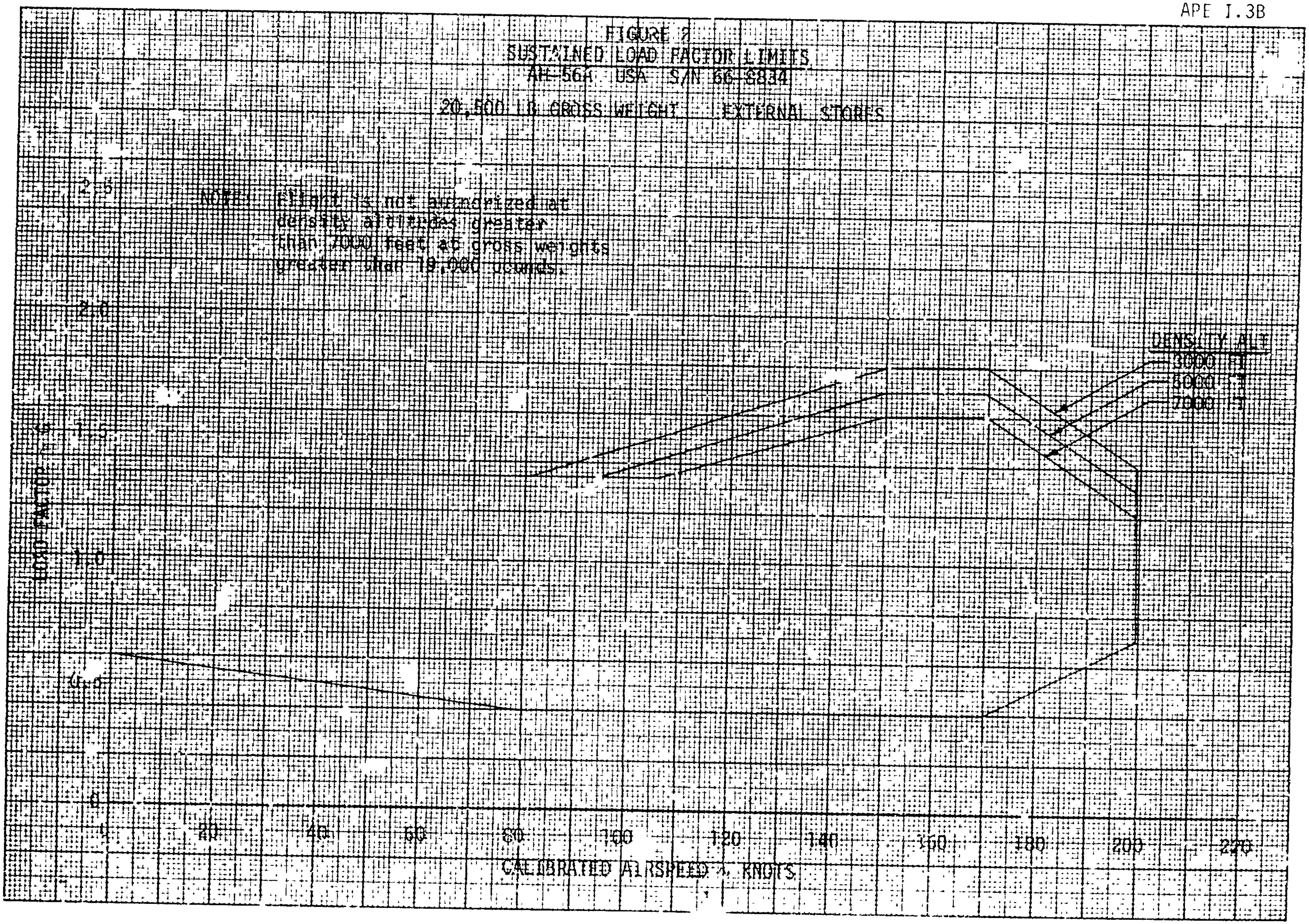


FIGURE 2
SUSTAINED LOAD FACTOR LIMITS
AH-56A USA S/N 66-8834
20,500 LB GROSS WEIGHT EXTERNAL STORES
NOTE: Flight is not authorized at density altitudes greater than 7000 feet at gross weights greater than 19,000 pounds.
DENSITY ALT
3000 FT
5000 FT
7000 FT
LOAD FACTOR ~ G
2.5
2.0
1.5
1.0
0.5
0
0
20
40
60
80
100
120
140
160
180
200
220
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 3
COLLECTIVE BLADE ANGLE LIMITS
AH-56A USA S/N 66-8834

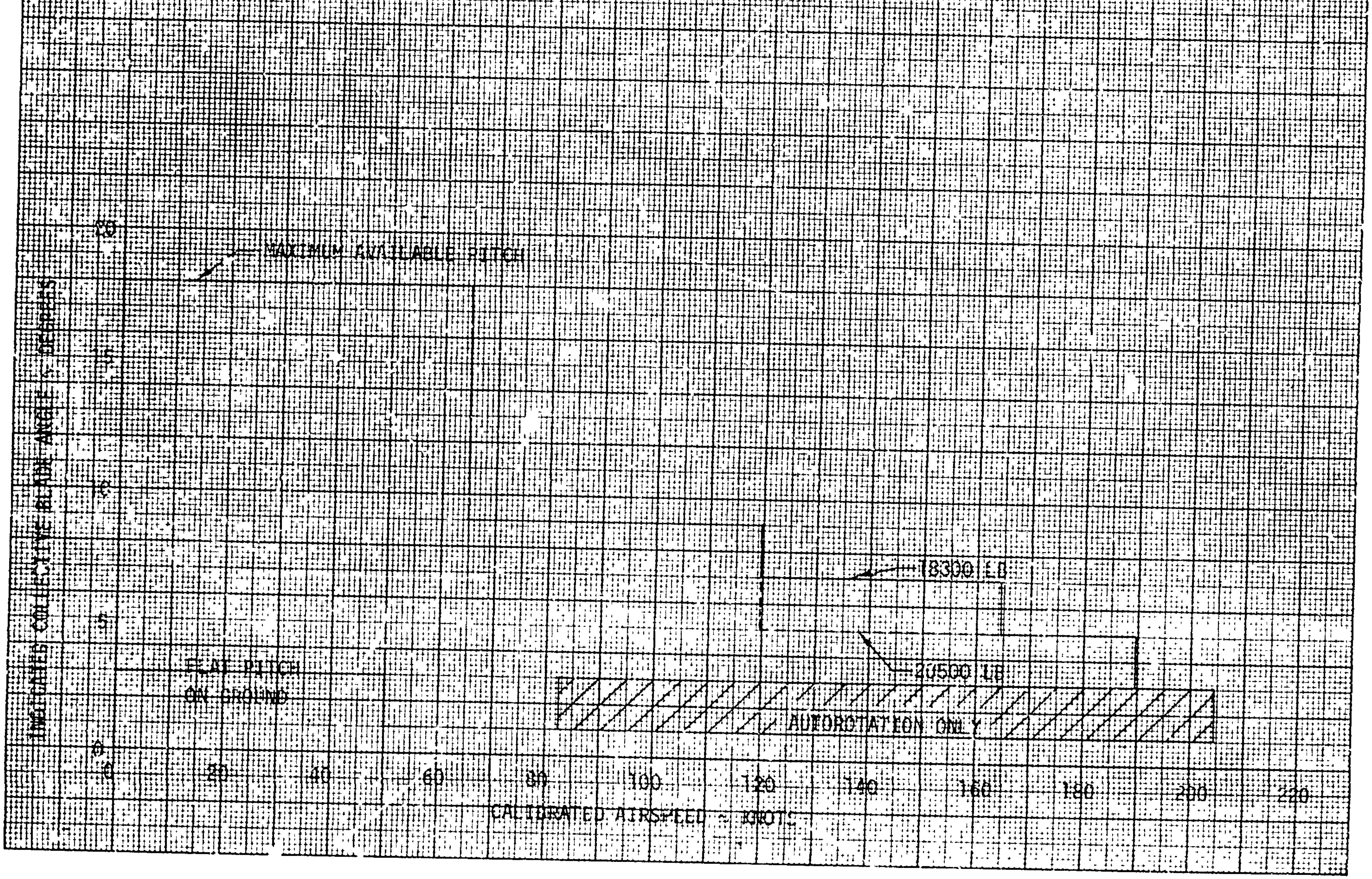
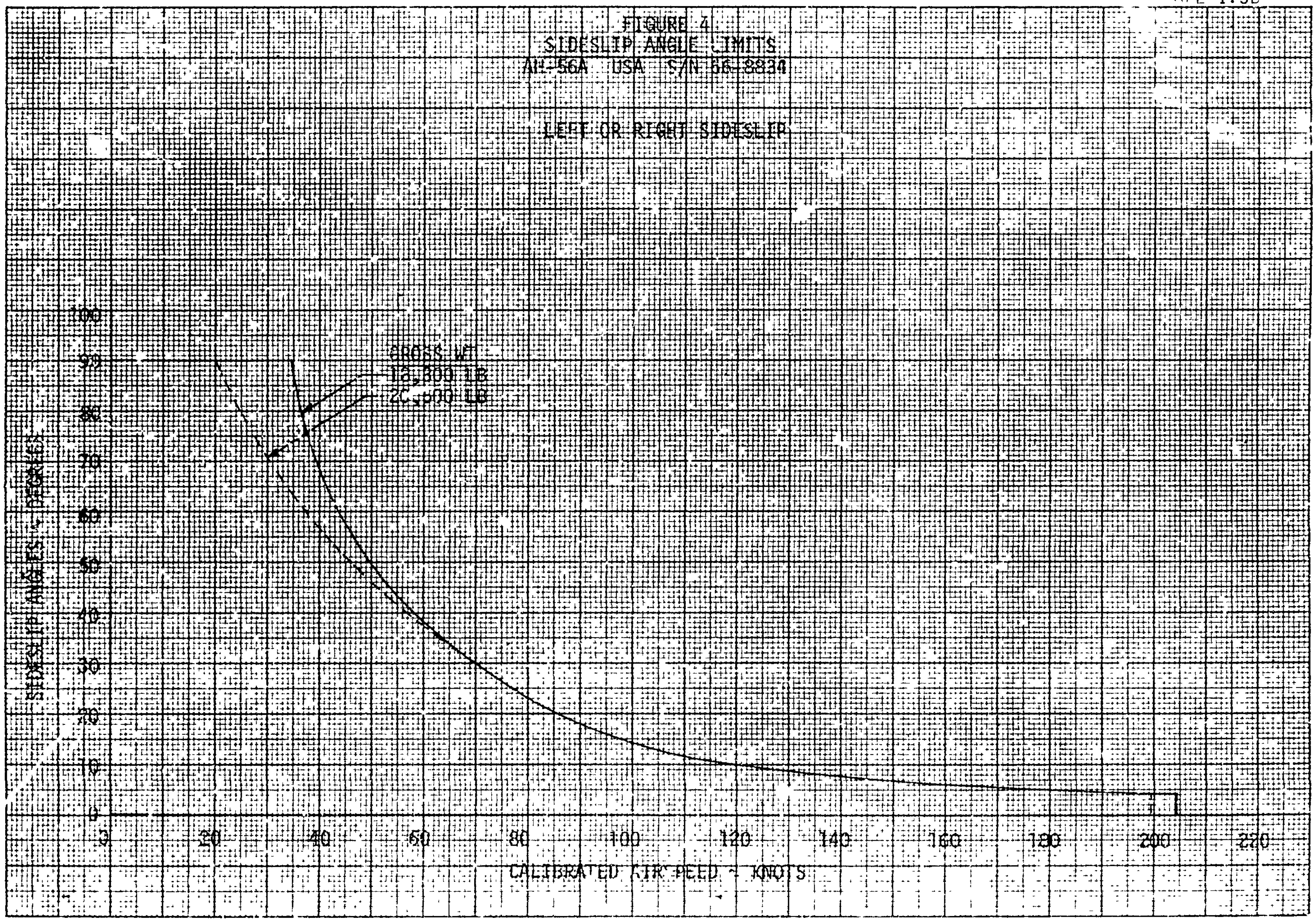
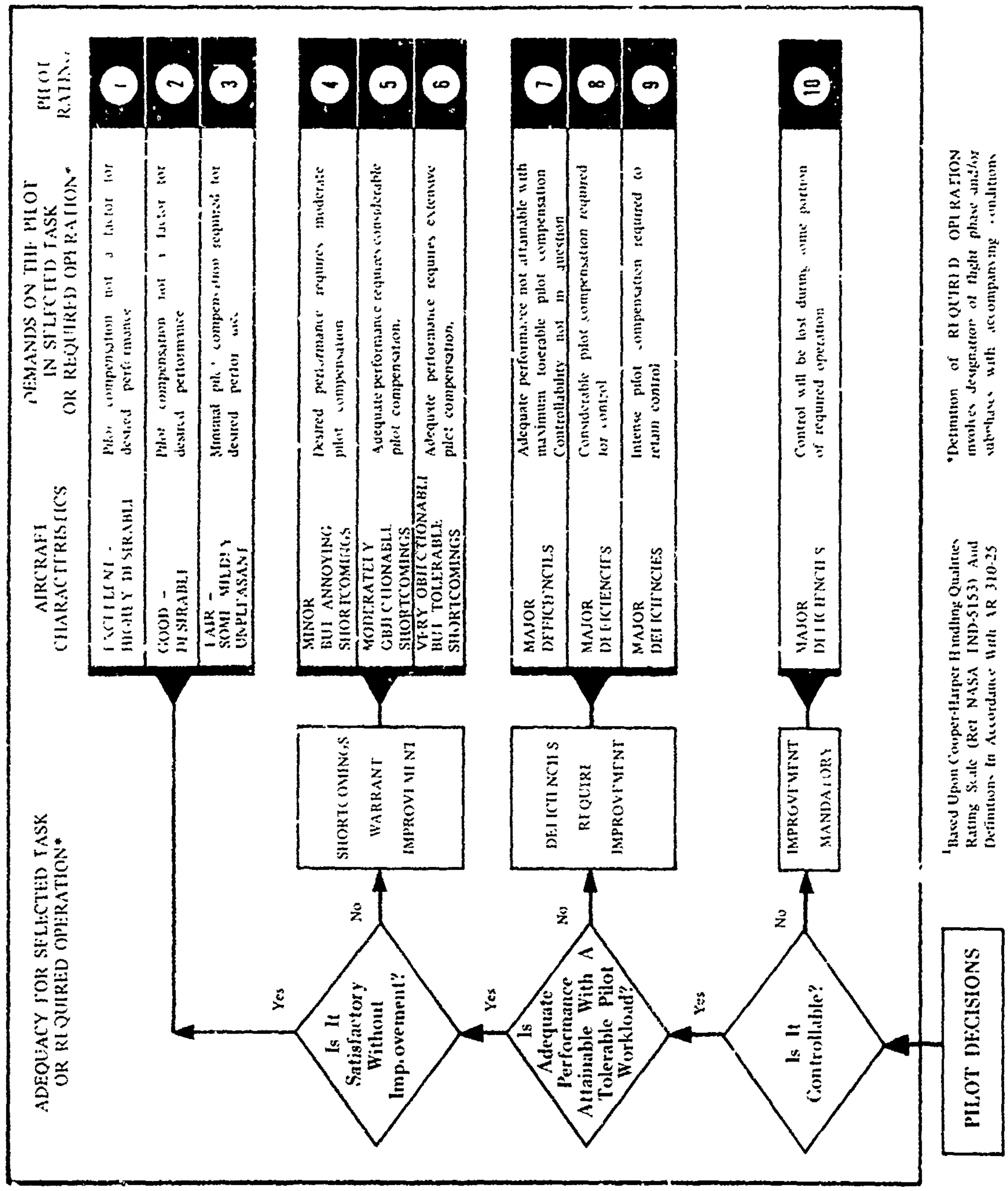


FIGURE 4
SIDESLIP ANGLE LIMITS
AH-56A USA S/N 66-8834

LEFT OR RIGHT SIDESLIP



APPENDIX D. HANDLING QUALITIES RATING SCALE



[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

APPENDIX E. TEST INSTRUMENTATION

1. Limited handling qualities test instrumentation was provided on the RDAT I test aircraft (S/N 66-8831). Since no quantitative data from the RDAT I test are presented in this report, the instrumentation list for S/N 66-8831 is not presented. Flight test instrumenation was installed in the APE I test aircraft (S/N 66-8834) prior to the start of this evaluation. This instrumentation provided data from the pilot panel (front seat), the photopanel, two oscillographs, and telemetry. The instrumentation was calibrated and maintained by the contractor throughout the test program. The following parameters were included in the instrumentation package:

Pilot Panel

Airspeed (boom)
Altitude (boom)
Rotor speed
Engine torque
Turbine inlet temperature
Longitudinal stick position
Lateral stick position
Pedal position
Collective stick position
Center-of-gravity normal acceleration
Angle of sideslip
Angle of attack
Pusher propeller Beta angle
Free air temperature
Gas producer speed
Fuel-used quantity
Time of day
Correlation counter

Photopanel

Airspeed (boom)
Altitude (boom)
Airspeed (ship's system)
Altitude (ship's system)
Free air temperature
Rotor speed
Gas generator speed
Power turbine speed
Fuel used
Fuel flow
Engine torque

Turbine inlet temperature
Time of day
Timer (10-second sweep)
Engine ignition box temperature
Pilot event
Propeller gearbox temperature
Correlation counter
Vertical speed
Transmission oil pressure
Engine oil pressure
Oil cooler differential pressure
Engine zone differential pressure
Total temperature

<u>Oscillograph #1</u>

Control positions.
- Longitudinal cyclic
- Lateral cyclic
- Collective

Control force:
- Longitudinal cyclic
- Lateral cyclic

Aircraft attitude:
- Pitch
- Roll

Aircraft rate:
- Pitch
- Roll

Center-of-gravity normal acceleration (filtered at 2 Hz)
Center-of-gravity lateral acceleration
Angle of attack
Angle of sideslip
Main rotor index
Pilot event
Tail drive shaft torque
Main drive shaft torque
Main rotor cyclic blade angle
Main rotor fixed hub flap bending at station 18
Main rotor fixed hub chord bending at station 18
Main rotor blade flap bending at station 174
Main rotor blade chord bending at station 174
Main rotor blade torsion at station 131.5
Main rotor shaft bending at zero degrees
Main rotor shaft bending at 90 degrees
Main rotor pitch link load
Main rotor gyro arm flap bending
Main rotor gyro drive torque
Swashplate collective position

Swashplate roll position
Swashplate pitch position
Pitch load below swashplate
Collective load below swashplate
Correlation counter

<u>Oscillograph #2</u>

Control positions:
    Pedal
    Pusher propeller blade angle (true)
Control force:
    Pedal
Aircraft rate:
    Yaw
Vibration:
    Pilot seat (vertical)
    Pilot seat (lateral)
    Copilot/gunner seat (vertical)
    Center of gravity (vertical)
    Center of gravity (lateral)
Tail rotor flap bending at station 5.2
Tail rotor chord bending at station 5.2
Tail rotor spindle support vertical bending
Tail rotor spindle support forward/aft bending
Tail rotor collective load
Tail rotor blade angle
Tail rotor shaft torque
Tail rotor index
Main rotor index
Correlation counter
Pilot event

<u>Telemetry</u> (A maximum of 18 parameters were transmitted for any one test. Different parameters were used, depending on the type of test. Output was provided on a bar scope, oscilloscope, and oscillograph, as well as being recorded on magnetic tape.)

Angle of sideslip
Angle of attack
Roll rate
Pitch rate
Yaw rate
Longitudinal cyclic position
Lateral cyclic position
Pedal position
Pusher propeller blade angle
Center-of-gravity normal acceleration

Main rotor chord bending at station 18
Main rotor chord bending at station 174
Main rotor chord bending, reactionless mode content
Gyro drive torque
Main rotor pitch link No. 1 load
Main rotor shaft bending at zero degrees
Collective control load (below swashplate)
Main rotor index
Main rotor blade angle
High-speed record indication and event
Main rotor blade torsion at station 131
Nose vertical acceleration at station 75

APPENDIX F. APE I.1 TEST DATA

This appendix includes the test data obtained during APE I.1. Data obtained during APE I.2, APE I.3A, and APE I.3B are presented in appendixes G, H and I, respectively. Various aircraft modifications were made during APE I. Details of these modifications can be found in appendix B. The aircraft modifications and configuration pertinent to each APE phase will be stated at the beginning of each data appendix and not necessarily repeated on the data figures. The rotor configuration for APE I.1 was a 4-degree forward sweep and a 2-degree 20-minute droop. The longitudinal control system incorporated a velocity gradient and a maneuver gradient with a 6-lb/g design effectiveness. The lateral control system incorporated a roll desensitizer and was rigged at 60-percent Phase III design effectiveness. No other stability or control augmentation devices were incorporated in the cyclic control system. The tests were conducted at the following nominal conditions: 4000-foot density altitude, 100-percent (246-rpm) rotor speed, 18,300-pound gross weight, station 300 cg location (defined as mid during APE I.1), and in the clean external configuration (no wing stores or store mounts). Trim airspeeds for each data point vary slightly from the presented airspeeds because of the difficulty and excessive time to obtain an exact trim speed (para 27). This variation is not considered to have significantly affected the data.

INDEX

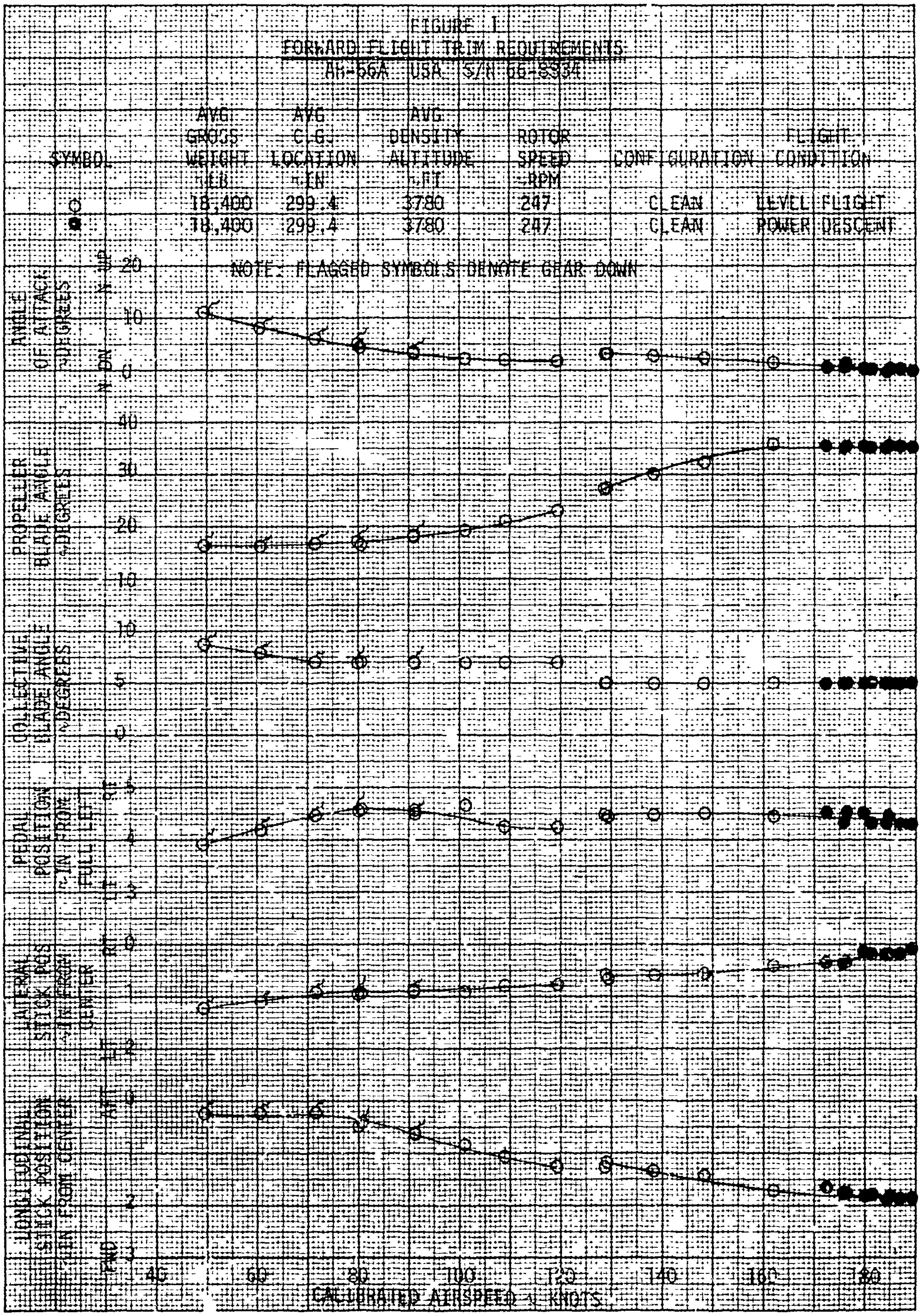


FIGURE 1
FORWARD FLIGHT TRIM REQUIREMENTS
AH-56A USA S/N 66-8834
SYMBOL
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
FLIGHT CONDITION
18,400 299.4 3780 247 CLEAN LEVEL FLIGHT
18,400 299.4 3780 247 CLEAN POWER DESCENT
NOTE: FLAGGED SYMBOLS DENOTE GEAR DOWN
ANGLE OF ATTACK ~DEGREES
N DN N UP
PROPELLER BLADE ANGLE ~DEGREES
COLLECTIVE BLADE ANGLE ~DEGREES
PEDAL POSITION ~IN FROM FULL LEFT
LATERAL STICK POS ~IN FROM CENTER
LONGITUDINAL STICK POSITION ~IN FROM CENTER
FWD AFT
CALIBRATED AIRSPEED ~ KNOTS

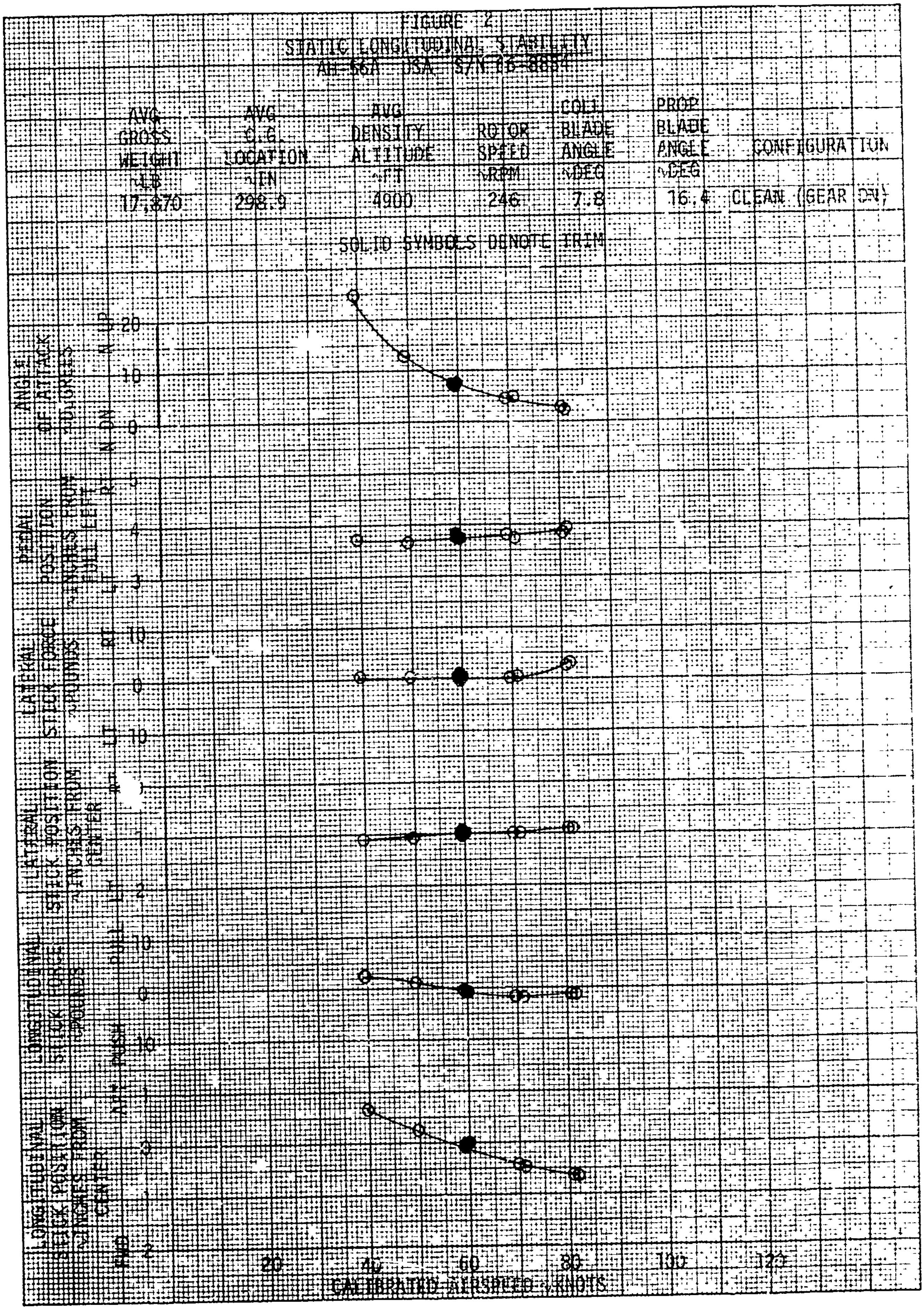


FIGURE 2
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB
17,870
AVG C.G. LOCATION ~IN
298.9
AVG DENSITY ALTITUDE ~FT
4900
ROTOR SPEED ~RPM
246
COLL BLADE ANGLE ~DEG
7.8
PROP BLADE ANGLE ~DEG
16.4
CONFIGURATION
CLEAN (GEAR DN)
SOLID SYMBOLS DENOTE TRIM
ANGLE OF ATTACK ~DEGREES
PEDAL POSITION ~INCHES FROM FULL LEFT
LATERAL STICK FORCE ~POUNDS
LATERAL STICK POSITION ~INCHES FROM CENTER
LONGITUDINAL STICK FORCE ~POUNDS
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER
CALIBRATED AIRSPEED ~KNOTS
20
40
60
80
100
120

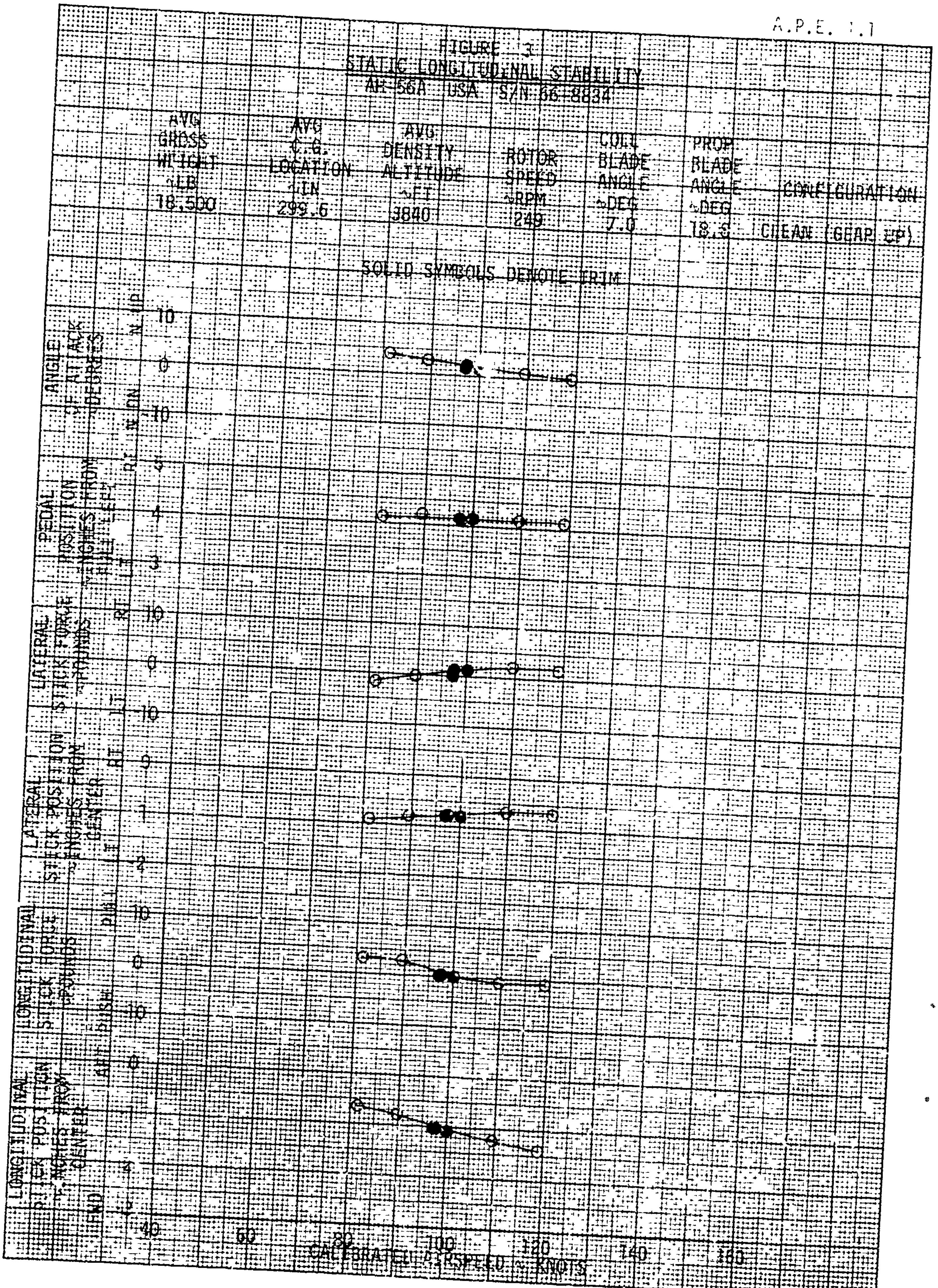
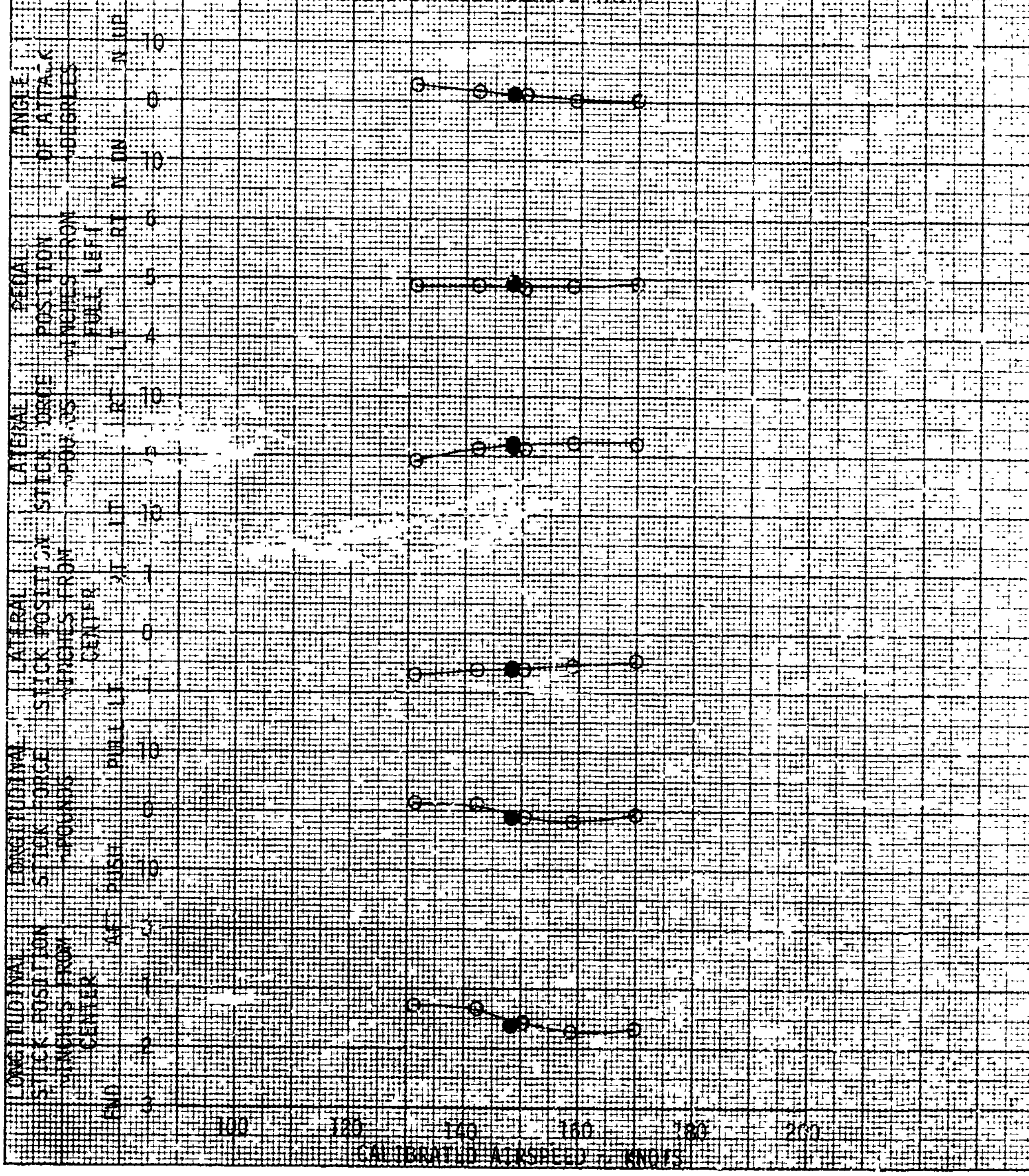

A.P.E. 1.1
FIGURE 3
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 18,500
AVG C.G. LOCATION ~IN 299.6
AVG DENSITY ALTITUDE ~FT 3840
ROTOR SPEED ~RPM 249
COLL BLADE ANGLE ~DEG 7.0
PROP BLADE ANGLE ~DEG 18.3
CONFIGURATION CLEAN (GEAR UP)
SOLID SYMBOLS DENOTE TRIM
ANGLE OF ATTACK ~DEGREES
N UP
N DN
PEDAL POSITION ~INCHES FROM FULL LEFT
RT
LT
LATERAL STICK FORCE ~POUNDS
LATERAL STICK POSITION ~INCHES FROM CENTER
LONGITUDINAL STICK FORCE ~POUNDS
PULL
PUSH
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER
AFT
FWD
CALIBRATED AIRSPEED ~ KNOTS
40
60
80
100
120
140
160

FIGURE 4
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,220	299.3	4030	246	5	29.8	CLEAN (GEAR UP)

SOLID SYMBOLS DENOTE TRIM



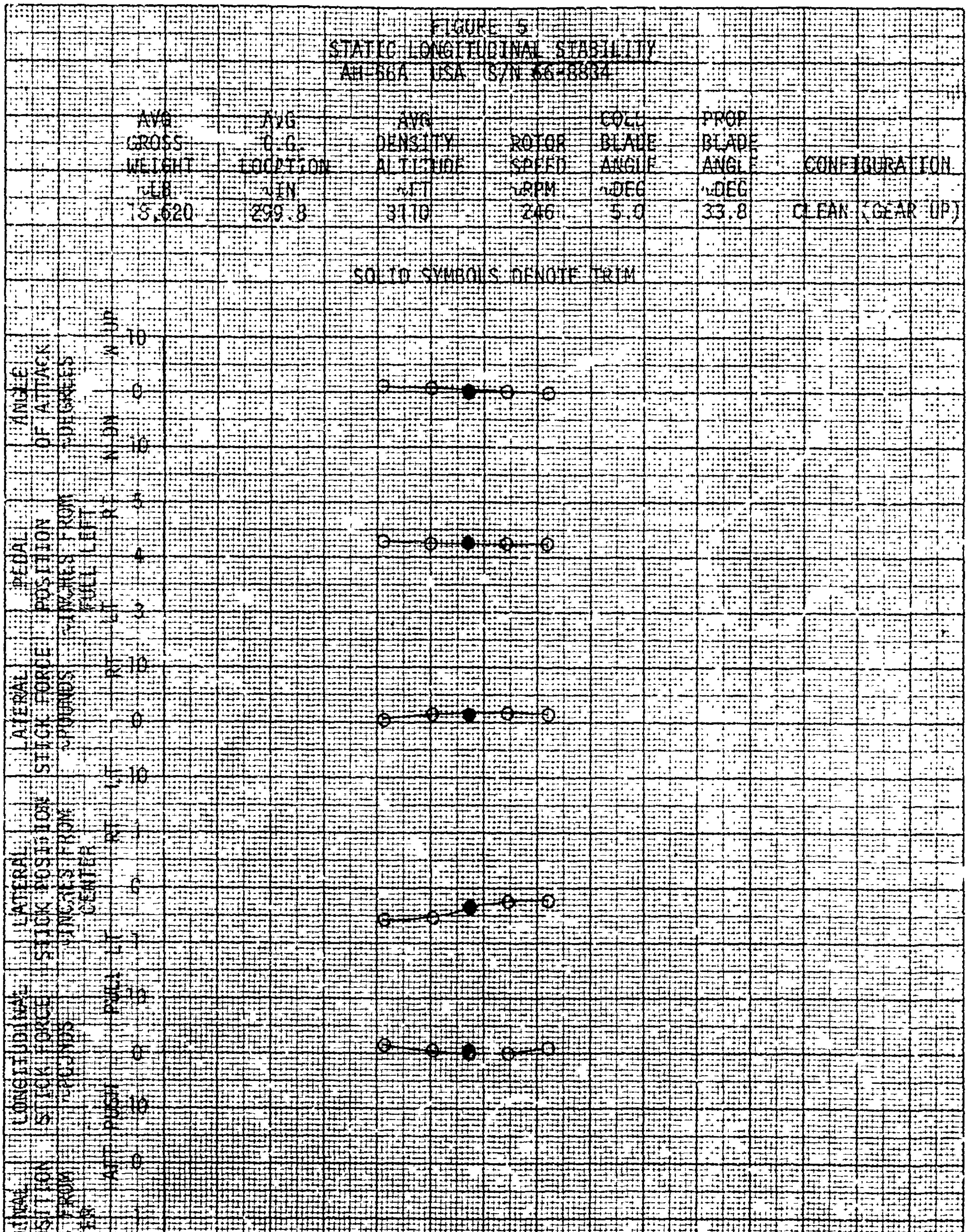


FIGURE 5
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 15,620
AVG C.G. LOCATION ~IN 299.8
AVG DENSITY ALTITUDE ~FT 3110
ROTOR SPEED ~RPM 246
COLL BLADE ANGLE ~DEG 5.9
PROP BLADE ANGLE ~DEG 33.8
CONFIGURATION CLEAN (GEAR UP)
SOLID SYMBOLS DENOTE TRIM
ANGLE OF ATTACK ~DEGREES
PEDAL POSITION ~INCHES FROM FULL LEFT
LATERAL STICK FORCE ~POUNDS
LATERAL STICK POSITION ~INCHES FROM CENTER
LONGITUDINAL STICK FORCE ~POUNDS
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER
A UP
A DN
RT
LT
PULL
PUSH
AFT
FWD
140
160
180
200
220
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 6
STATIC LATERAL DIRECTIONAL STABILITY
AH-56A USA S/N 66-8824

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
17,850	259	3900	245	9.46	14.38	69	CLEAN (GEAR ON)

SOLID SYMBOLS DENOTE TRIM

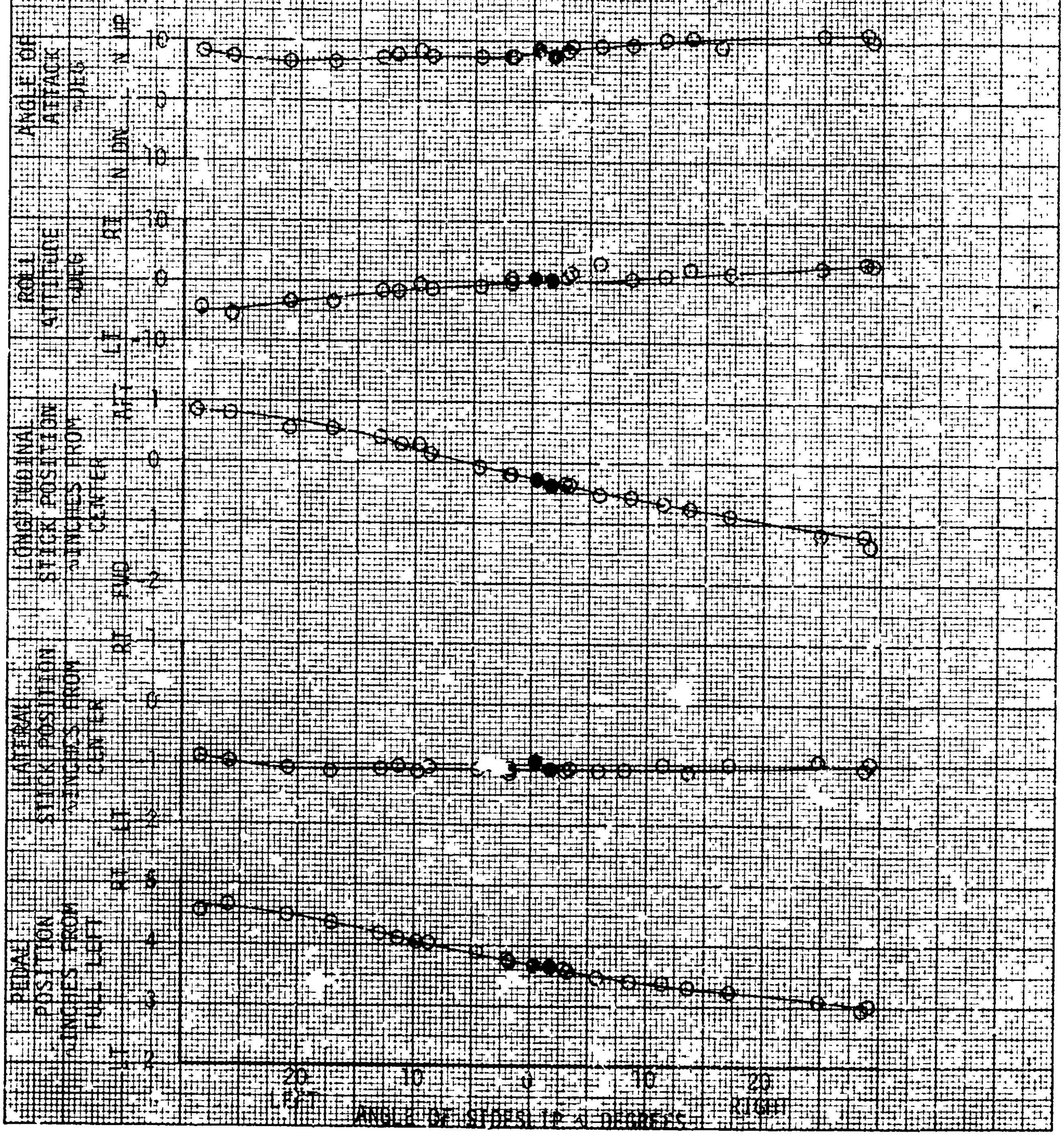


FIGURE 7
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
18,450	299.6	4540	248	7.0	18.5	99	CLEAN (GEAR UP)

SOLID SYMBOLS DENOTE TRIM

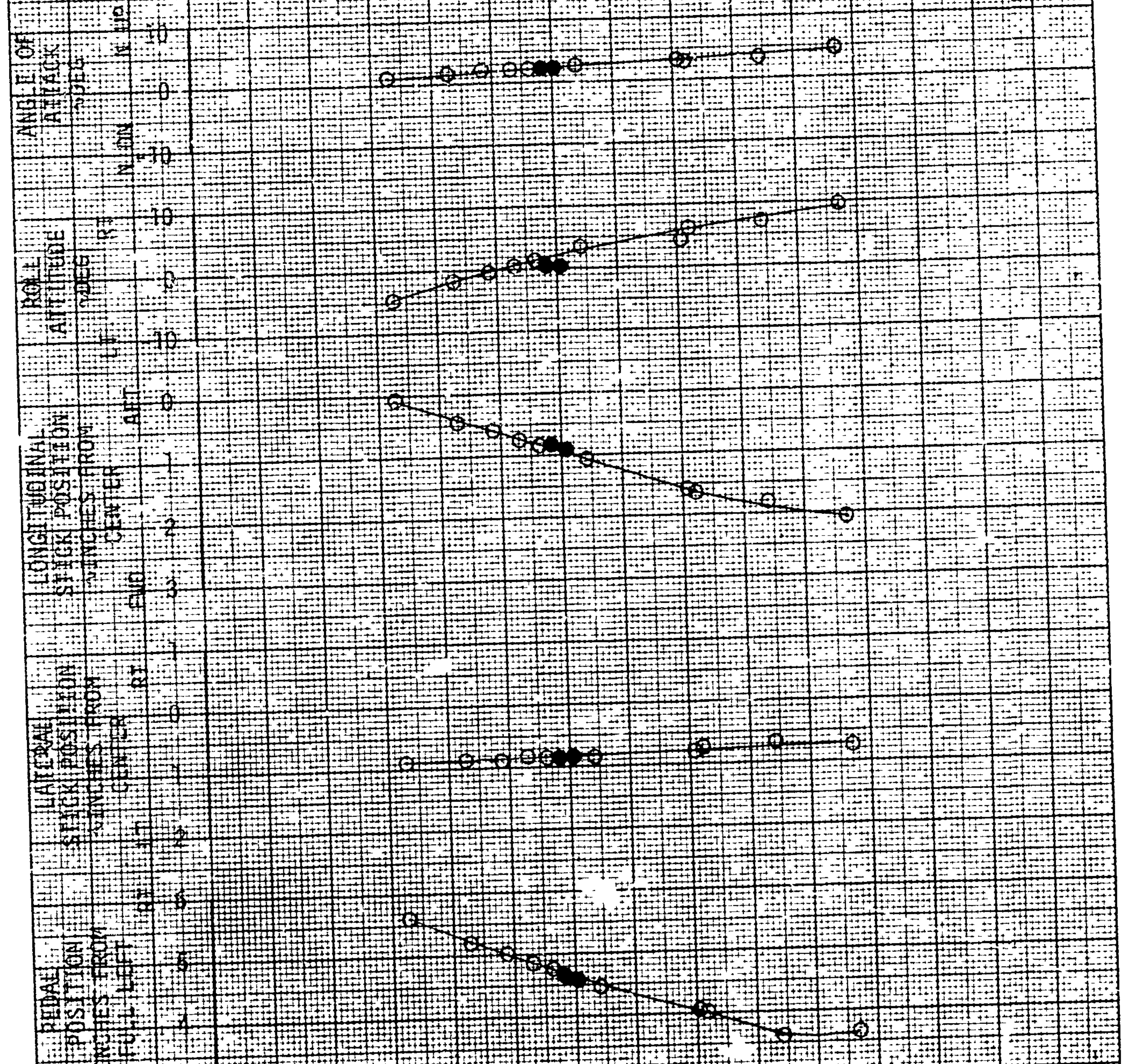


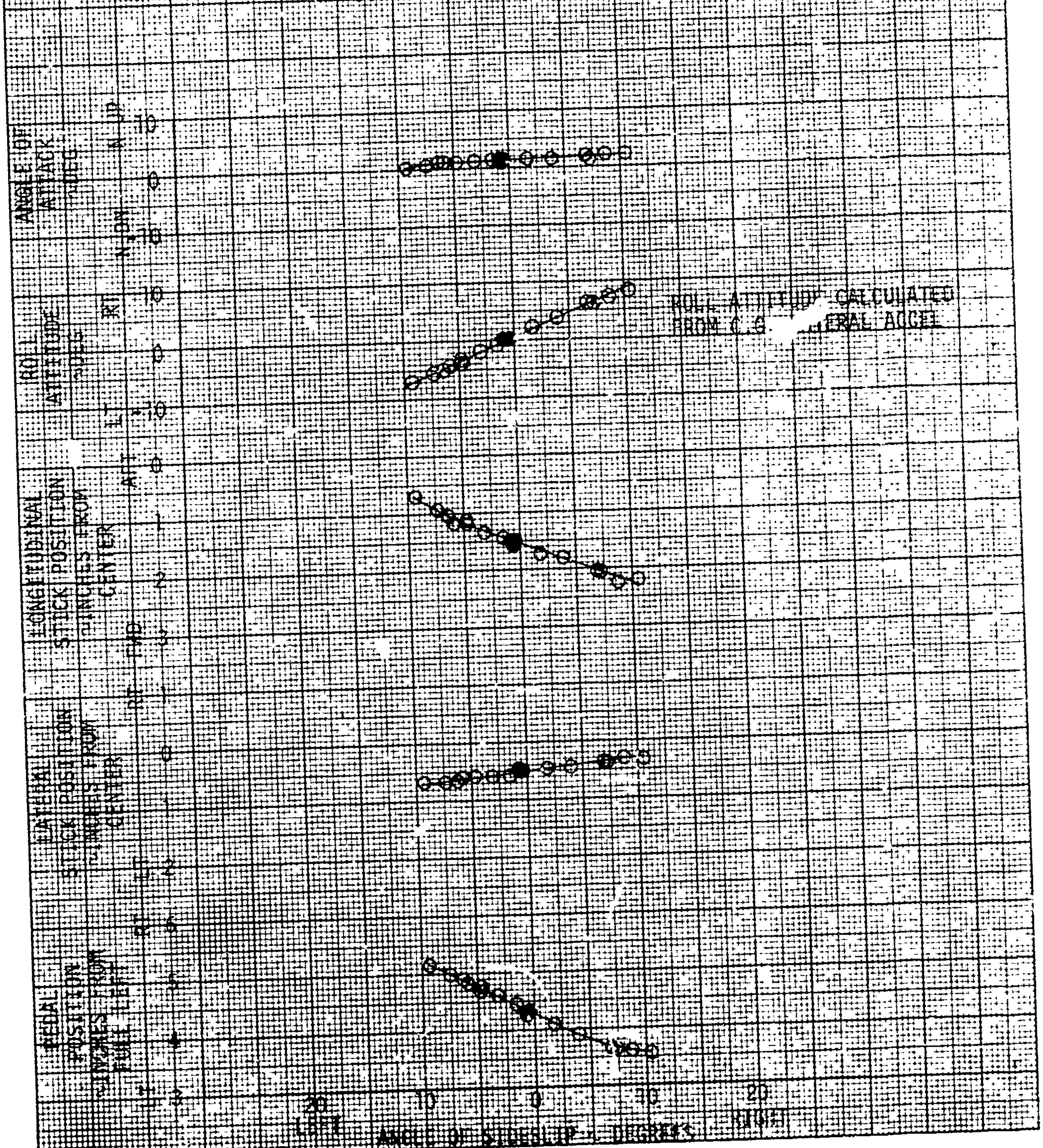
FIGURE 8

STATIC LATERAL-DIRECTIONAL STABILITY

AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
18,110	293.3	4430	246	5.1	31.4	150	CLEAN (GEAR UP)

SOLID SYMBOLS DENOTE TRIM

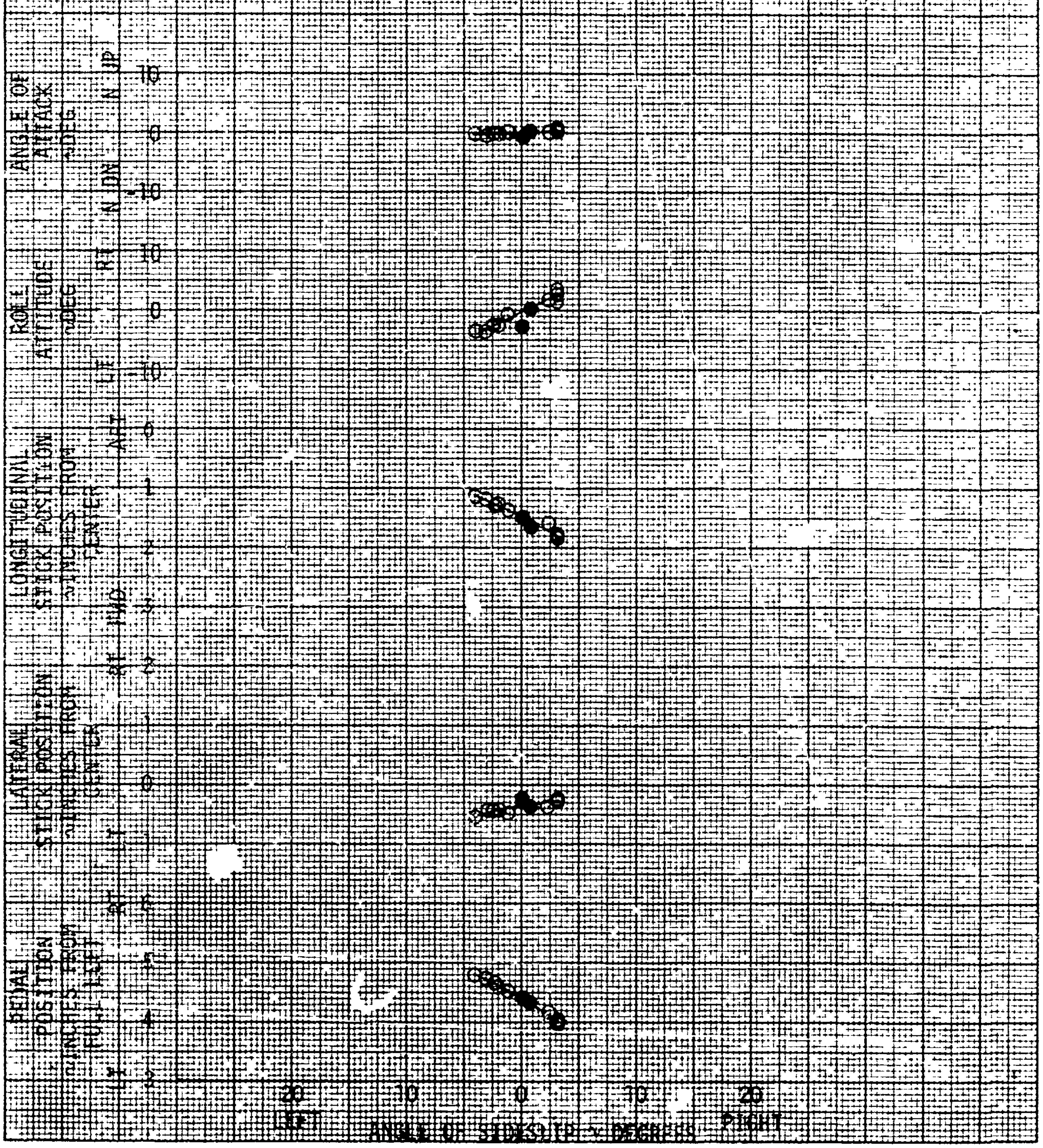


A.P.E. 1.1

FIGURE 9
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
18,390	299.7	3140	246	5.0	33.7	178	CLEAN (GEAR UP)

SOLID SYMBOLS DENOTE TRIM



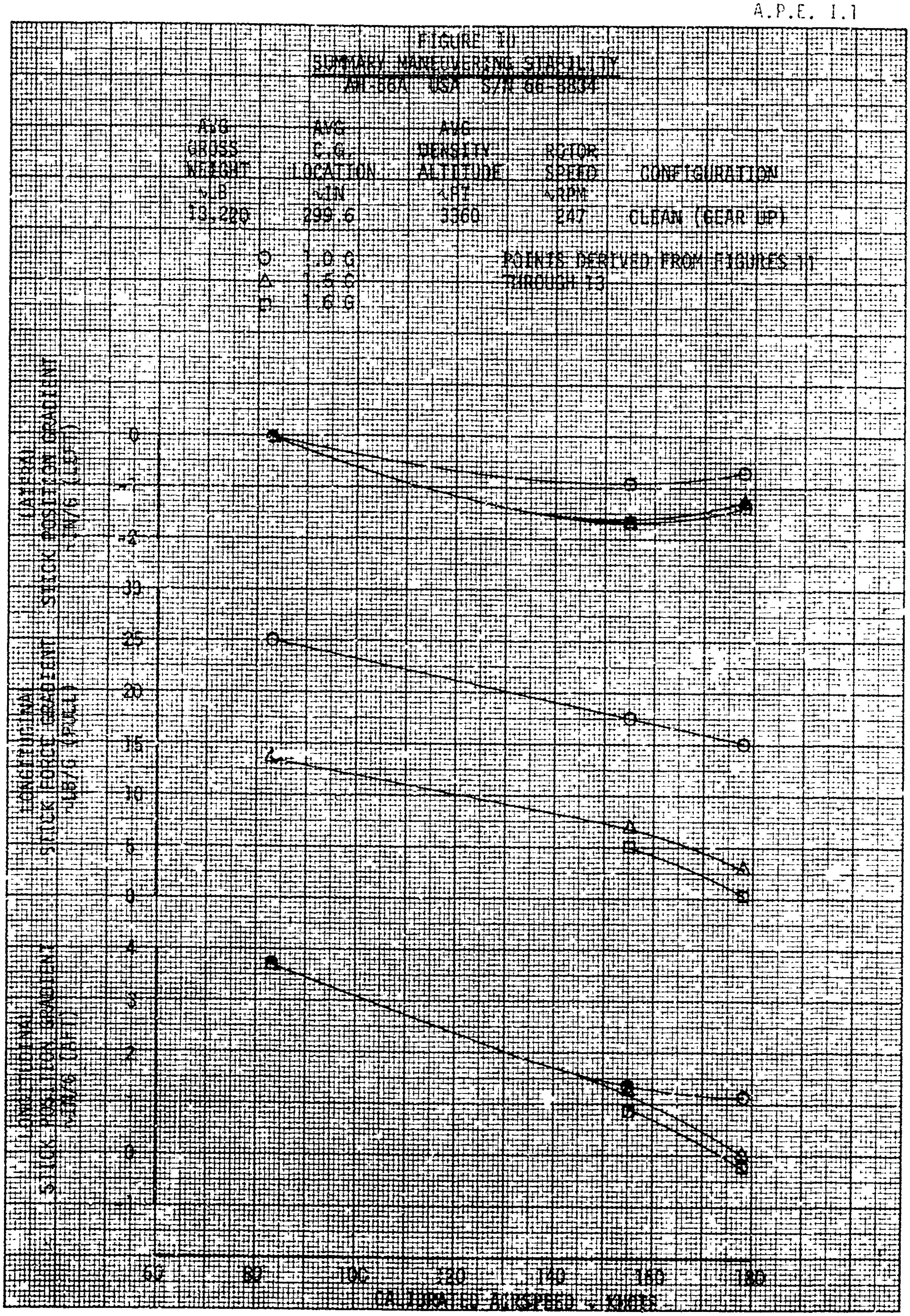


FIGURE 10
SUMMARY MANEUVERING STABILITY
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
13,220
299.6
3360
247
CLEAN (GEAR UP)
1.0 G
1.5 G
1.6 G
POINTS DERIVED FROM FIGURES 11 THROUGH 13
LONGITUDINAL
STICK POSITION GRADIENT
LATERAL
STICK FORCE GRADIENT
STICK POSITION GRADIENT
CALIBRATED AIRSPEED ~ KNOTS
60
80
100
120
140
160
180

FIGURE 11
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
○	18,070	299.7	3280	247	7.2	96	CLEAN (GEAR UP)
□	18,120	299.6	3380	248	7.2	80	CLEAN (GEAR UP)

○ LEFT WIND UP TURN
□ RIGHT WIND UP TURN

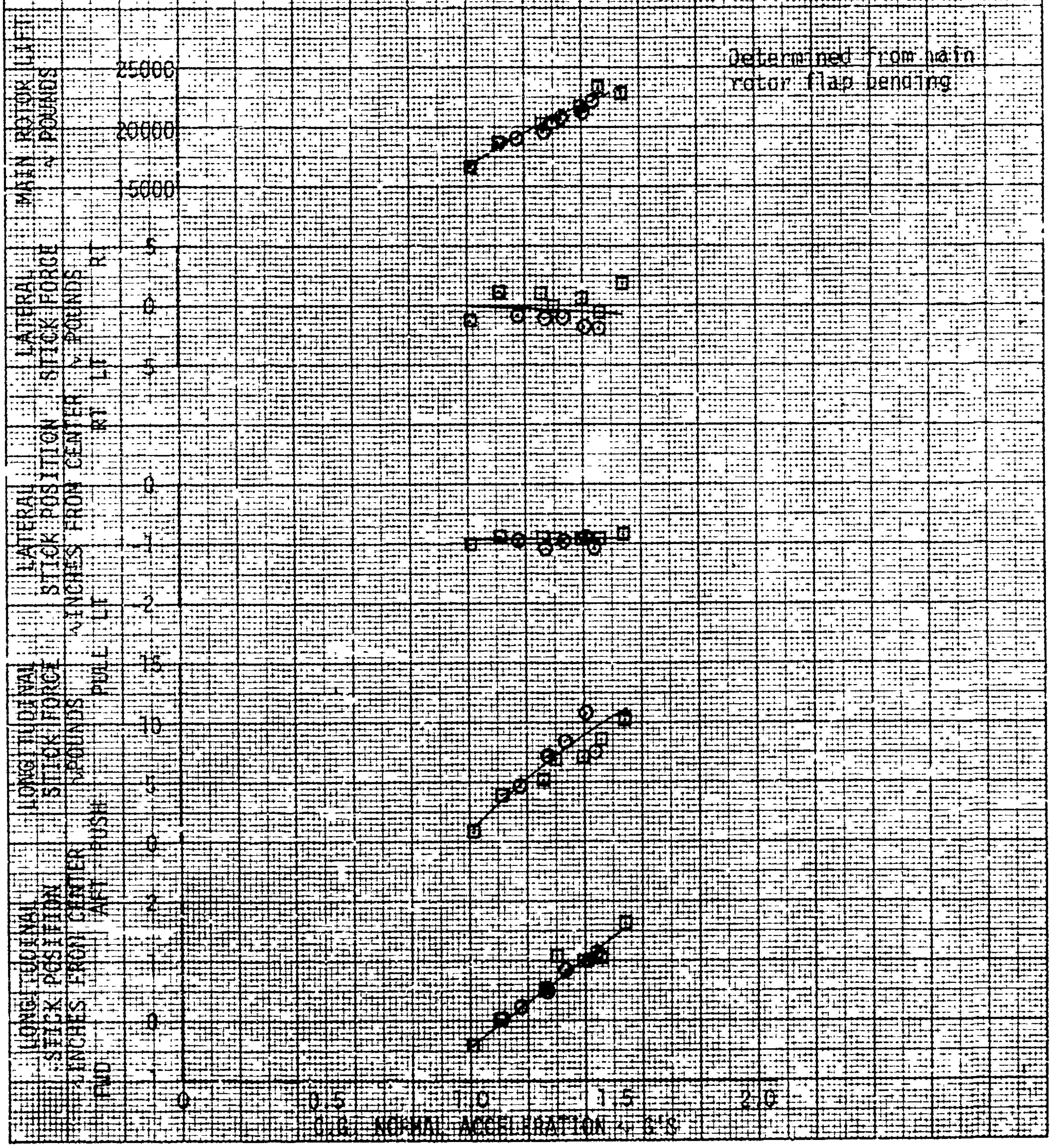


FIGURE 12
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
○	18,200	299.4	3880	247	6.0	154	CLEAN (GEAR UP)
□	18,648	299.5	4200	248	6.1	156	CLEAN (GEAR UP)

○ LEFT WIND UP TURN
⊕ RIGHT WIND UP TURN

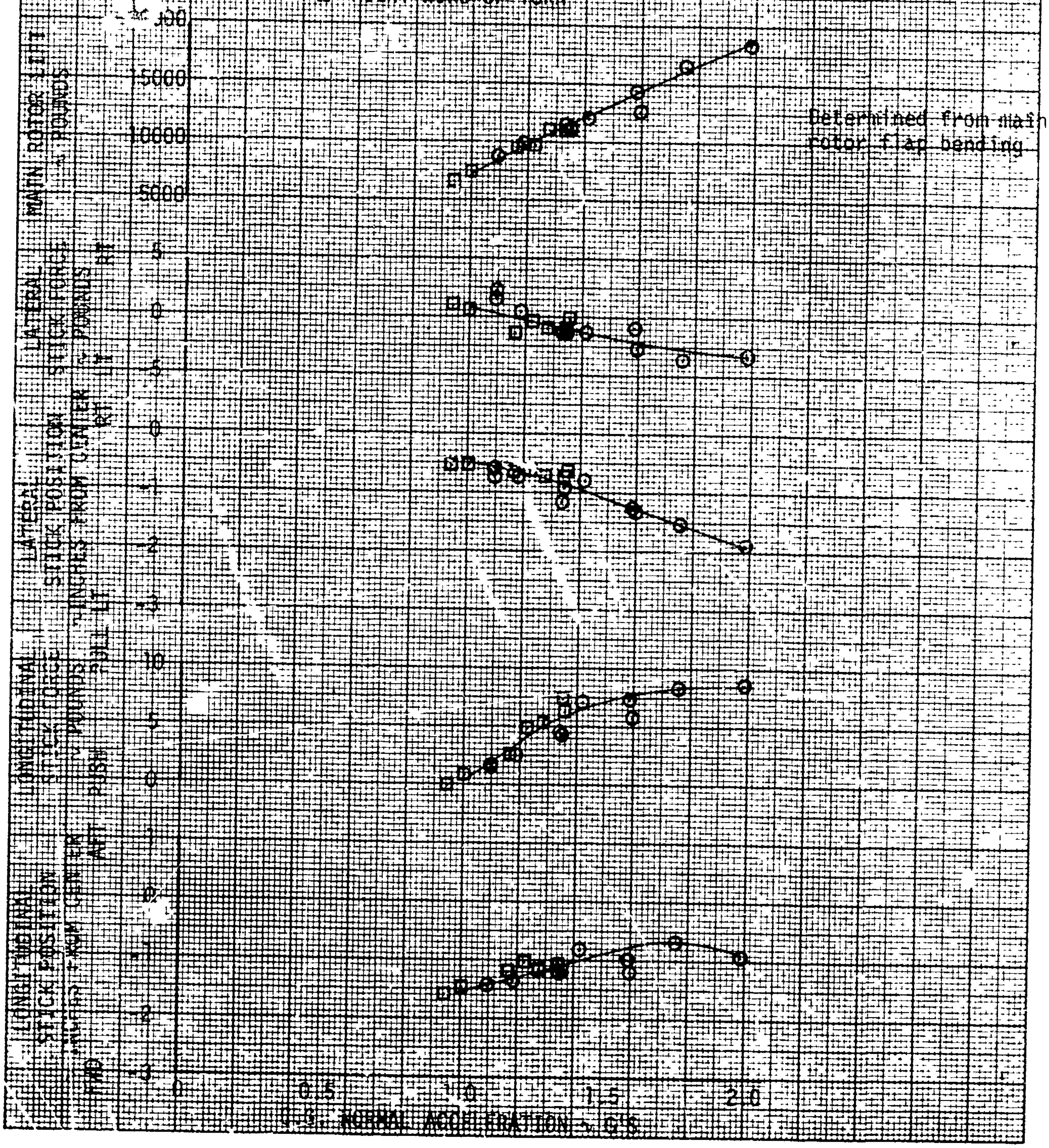


FIGURE 13
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
○	18,180	299.6	3090	247	5.1	178	CLEAN (GEAR UP)
□	18,040	299.6	3340	247	5.1	178	CLEAN (GEAR UP)

○ LEFT WIND UP TURN
□ RIGHT WIND UP TURN

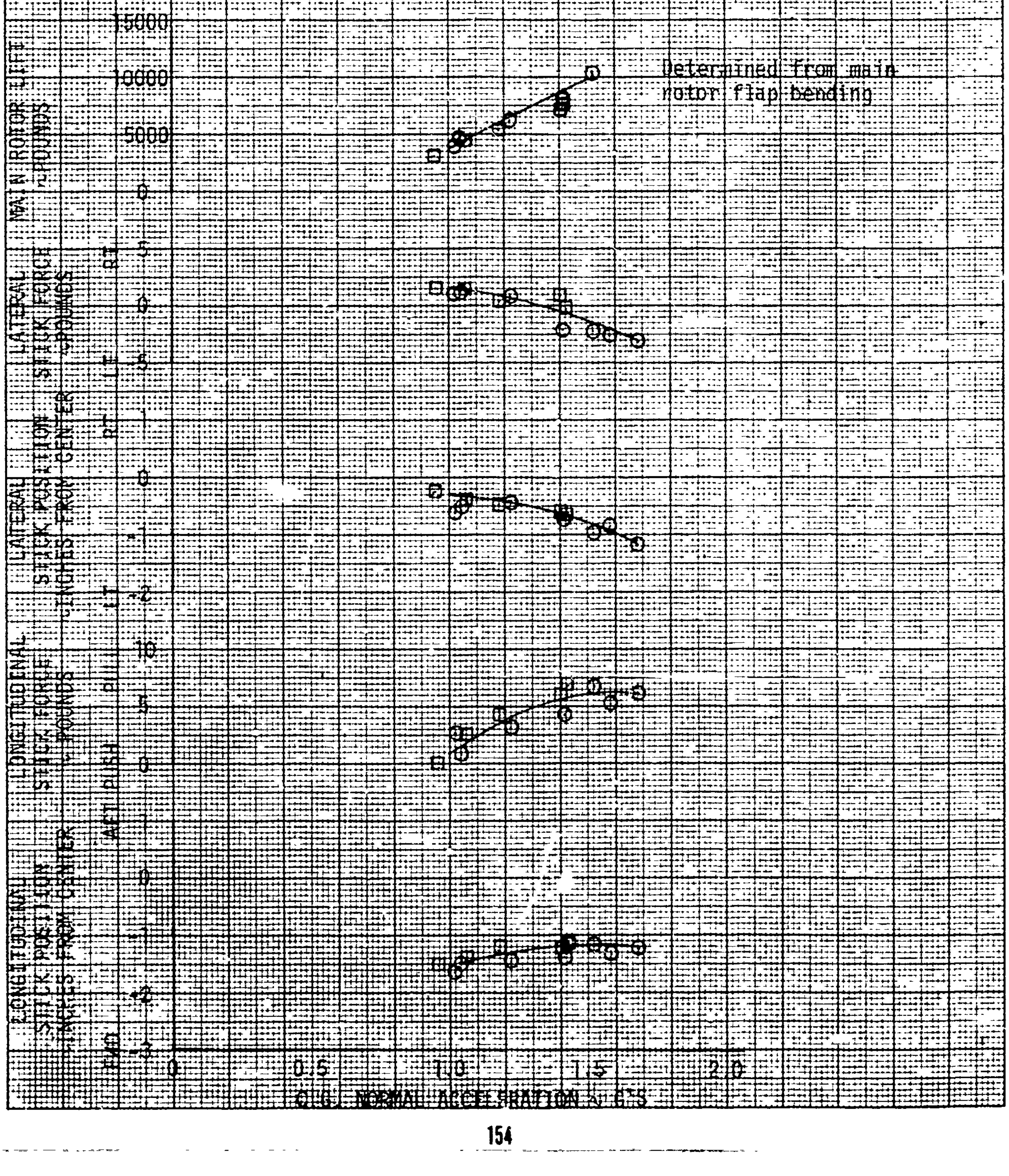


FIGURE 14

LEFT WIND-UP TURN

AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,480	299.7	4750	250	165	5.0	33	CLEAN (GEAR UP)

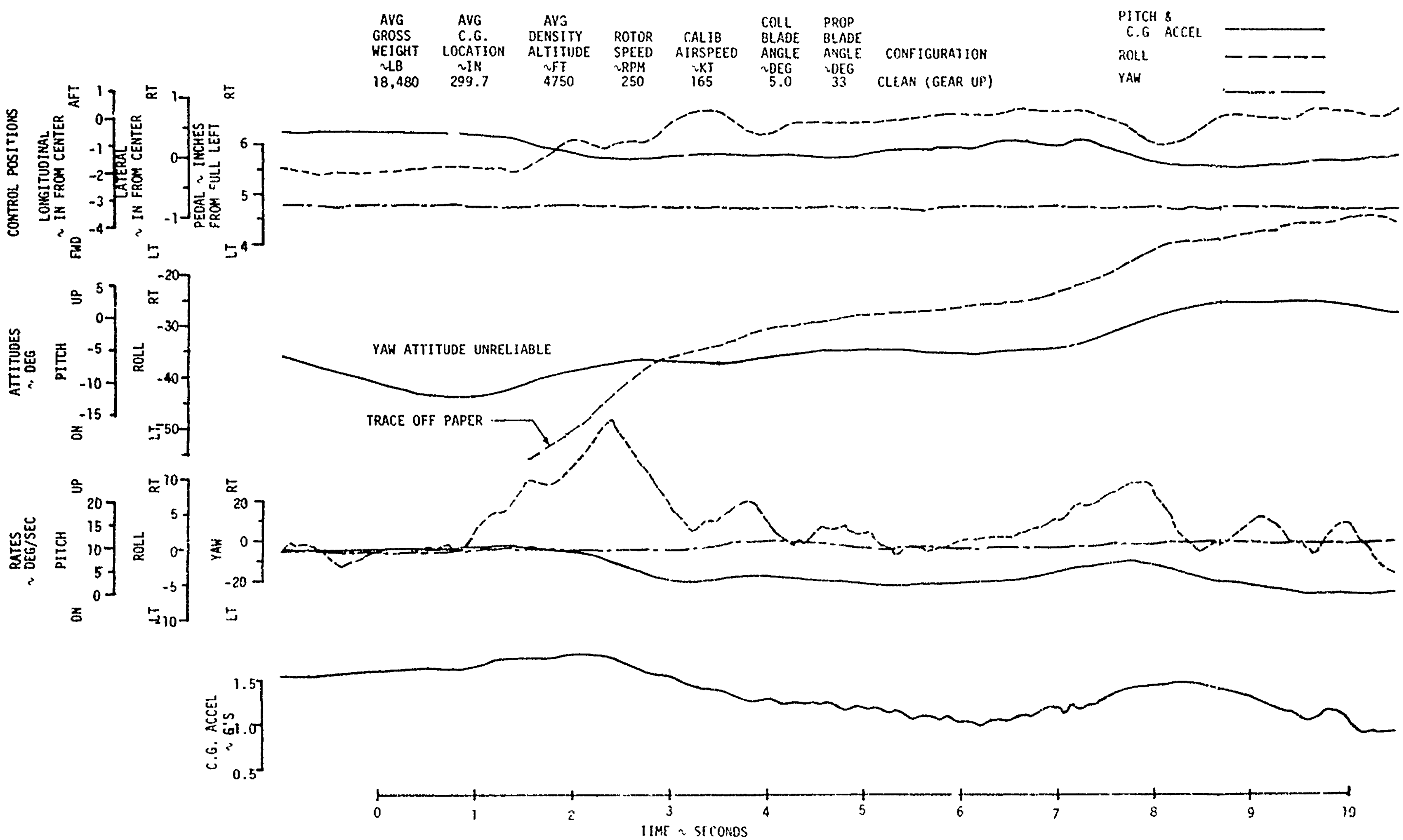


FIGURE 15
SUMMARY LONGITUDINAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
18,400	299.5	4030	247	CLEAN	LEVEL FLIGHT
18,600	299.8	-440	247	CLEAN	HOVER (60-75 FT)

POINTS DERIVED FROM FIGURES 16 THROUGH 19
TAILS DENOTE GEAR DOWN

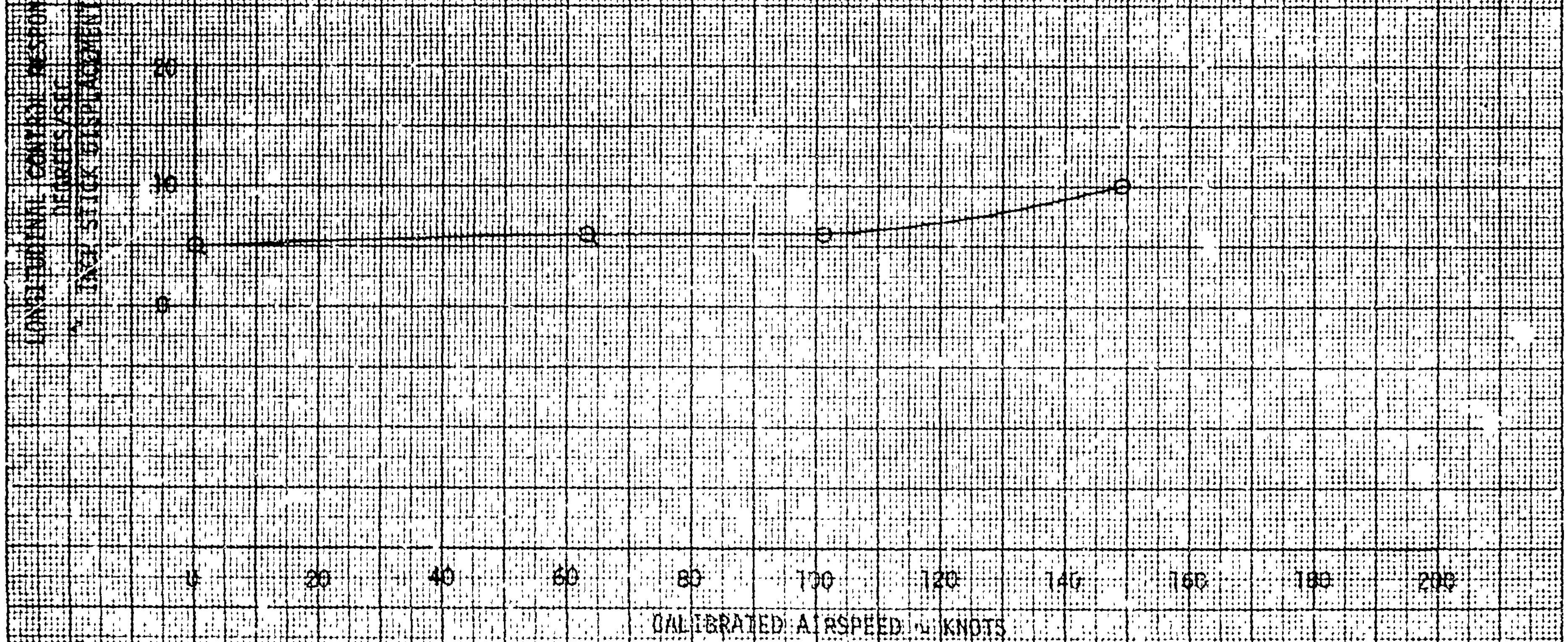
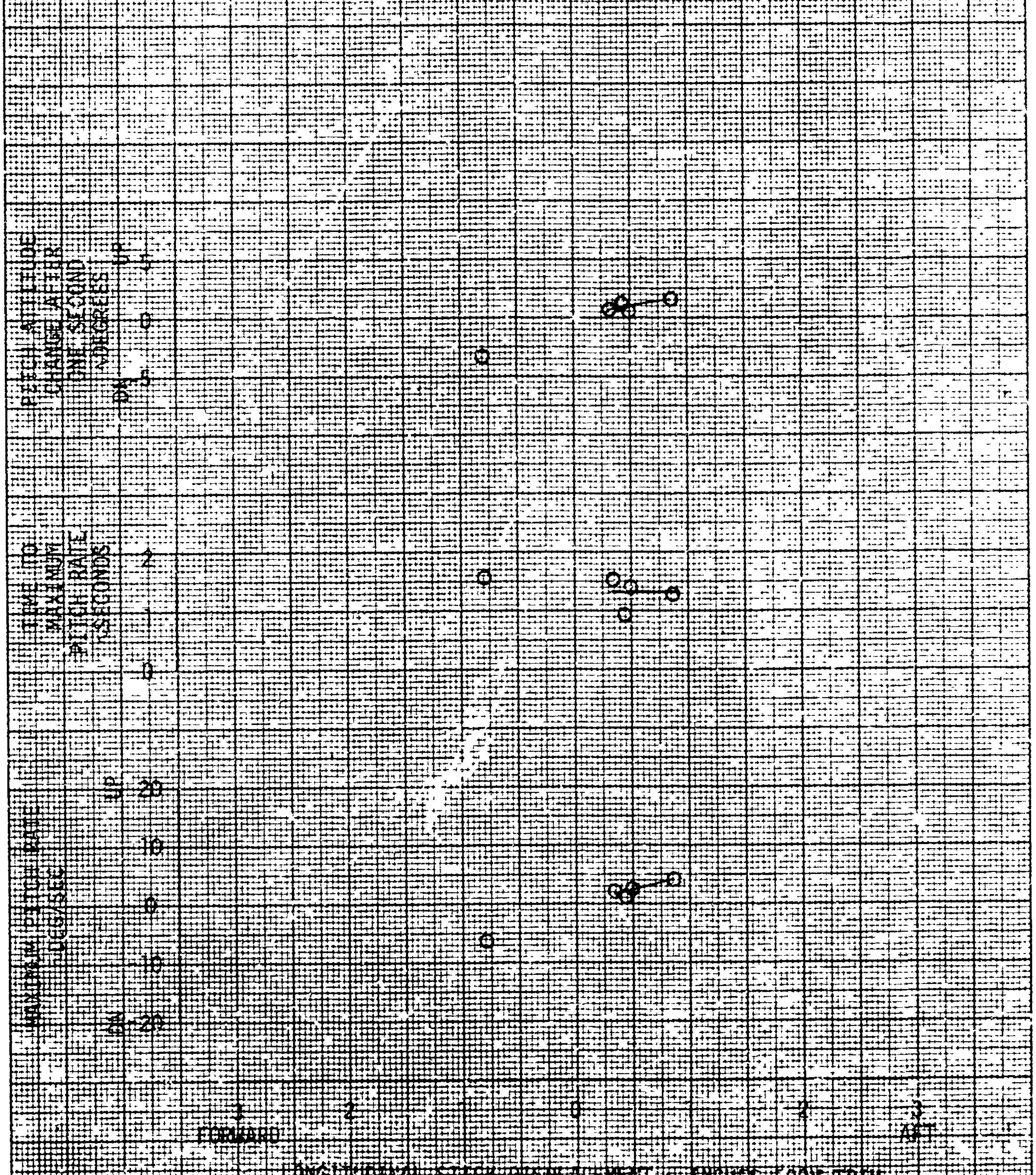


FIGURE 16
LONGITUDINAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,600	299.8	-440	247	11.7	HOVER (IGE-15 FT)	CLEAN (GEAR DN)



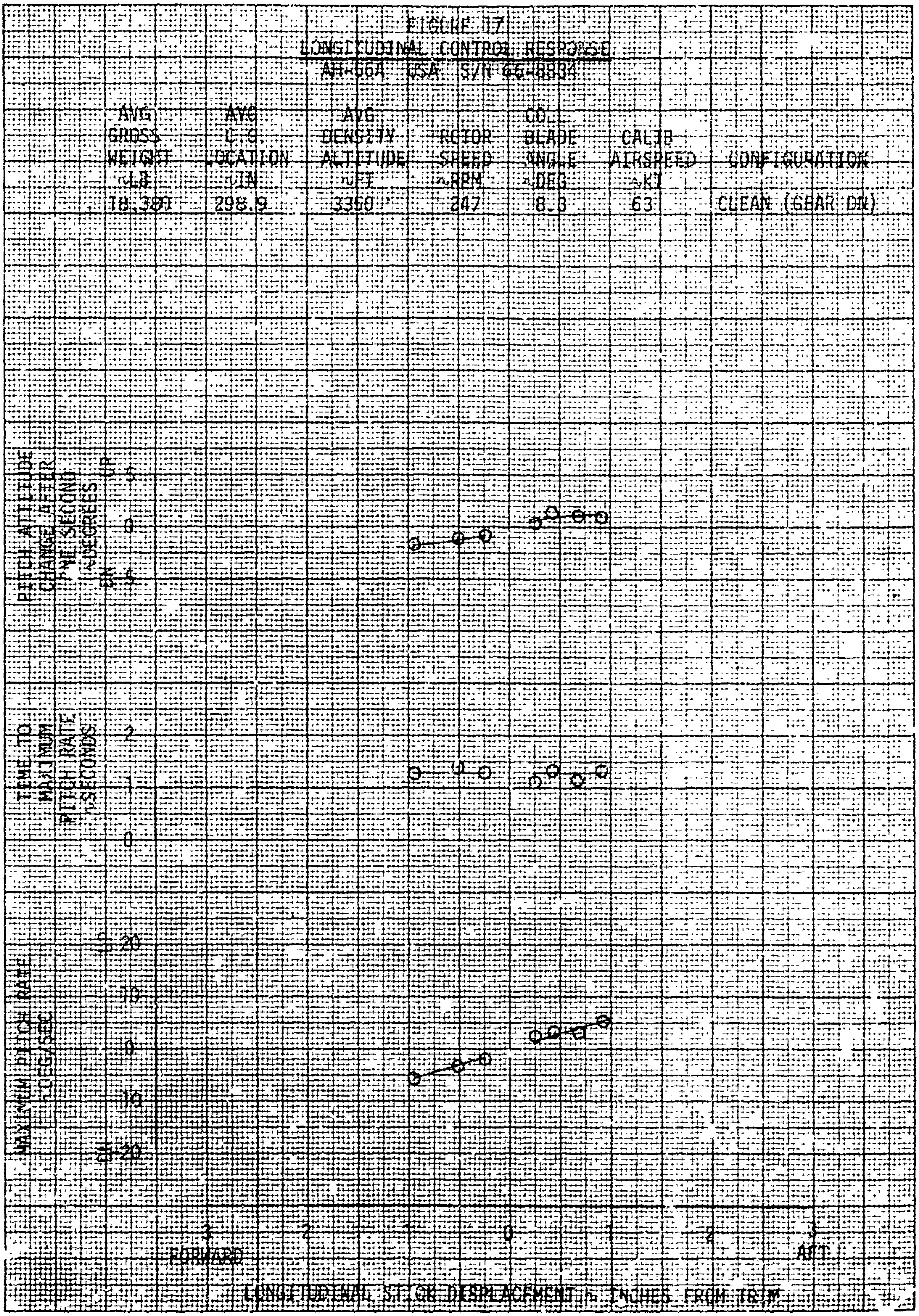


FIGURE 17
LONGITUDINAL CONTROL RESPONSE
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL BLADE ANGLE ~DEG
CALIB AIRSPEED ~KT
CONFIGURATION
18,380
298.9
3350
247
8.3
63
CLEAN (GEAR DN)
PITCH ATTITUDE CHANGE AFTER ONE SECOND ~DEGREES
UP
DN
5
0
5
TIME TO MAXIMUM PITCH RATE ~SECONDS
2
1
0
MAXIMUM PITCH RATE ~DEG/SEC
UP 20
10
0
10
DN 20
3
2
1
0
1
2
3
FORWARD
AFT
LONGITUDINAL STICK DISPLACEMENT ~ INCHES FROM TRIM

FIGURE 18
LONGITUDINAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,510	299.8	4140	246	7.0	10	CLEAN (GEAR UP)

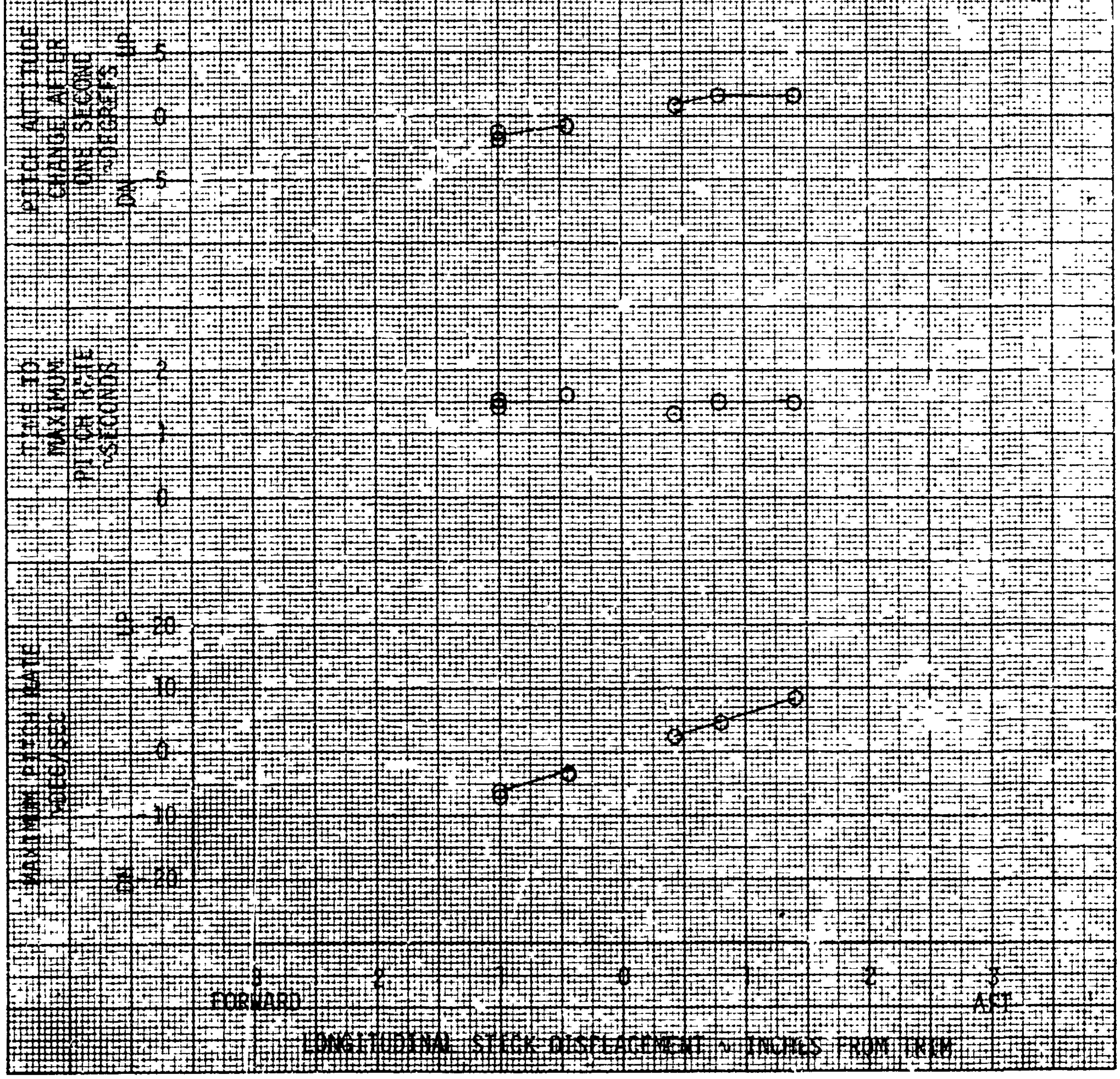
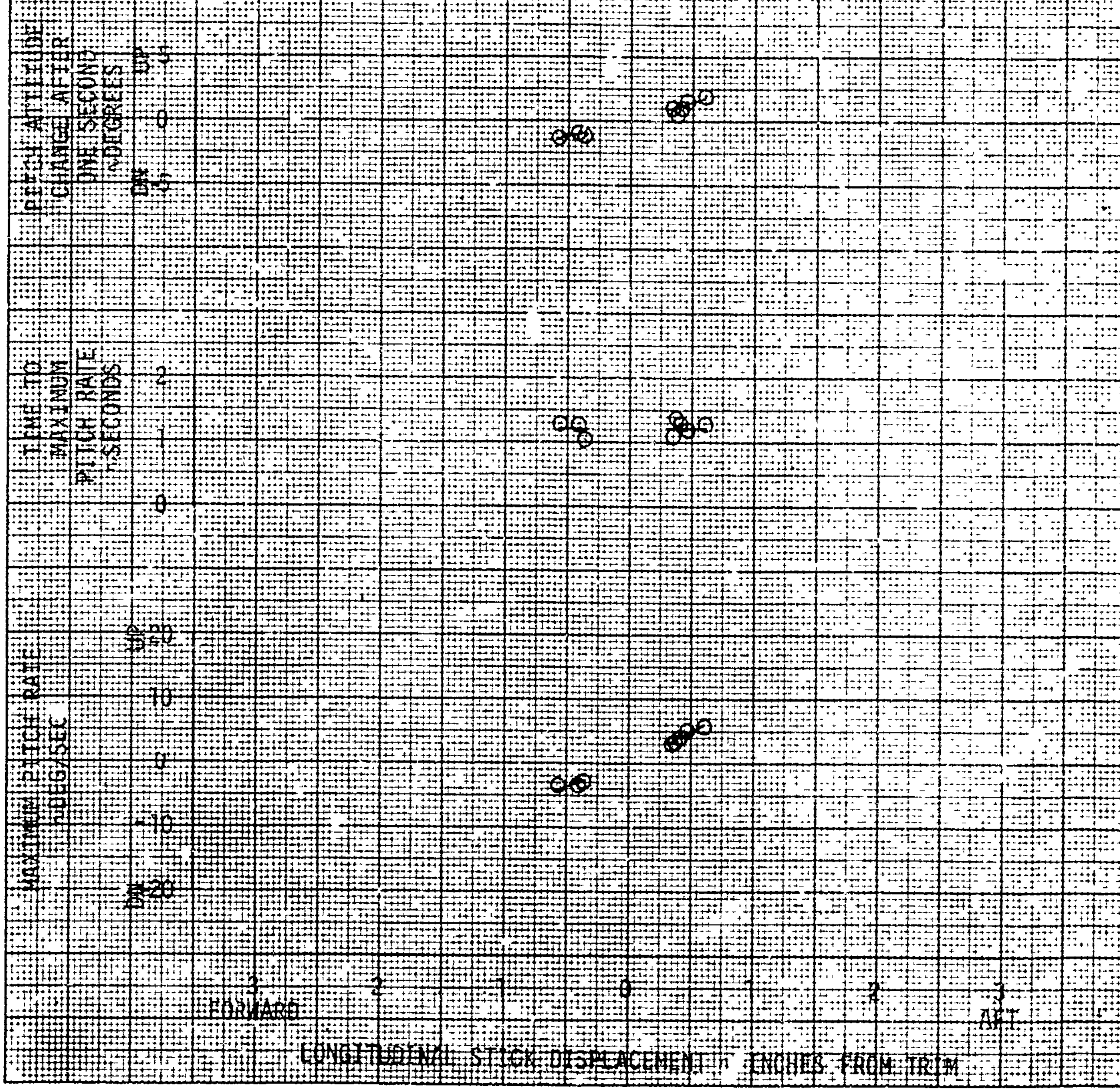


FIGURE 15
LONGITUDINAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,300	299.7	4600	248	5.4	149	CLEAN (GEAR UP)



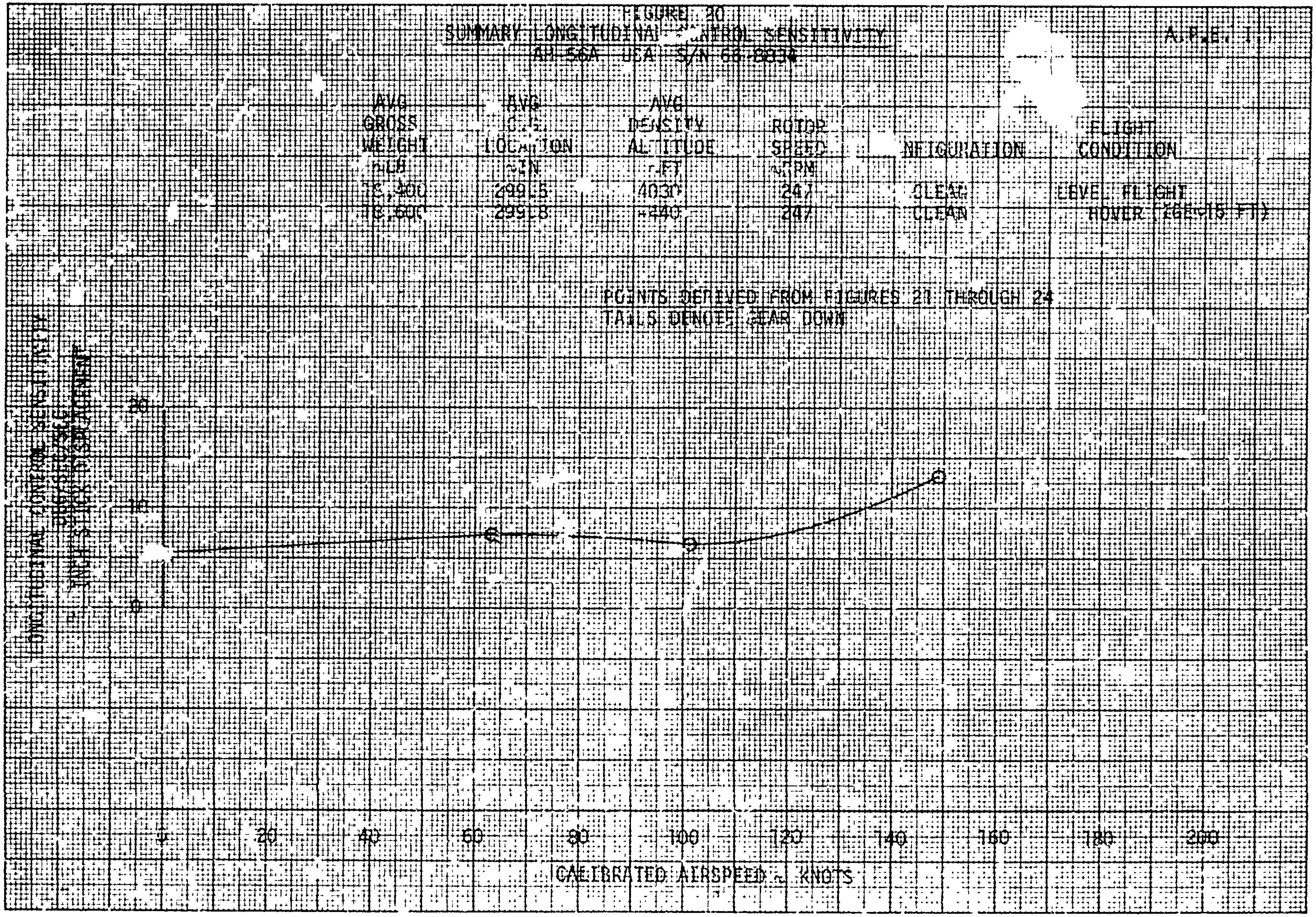


FIGURE 30
SUMMARY LONGITUDINAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834
A.P.E. 1.1
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
FLIGHT CONDITION
299.5
4030
247
CLEAN
LEVEL FLIGHT
299.8
247
CLEAN
HOVER (IGE-15 FT)
POINTS DERIVED FROM FIGURES 21 THROUGH 24
TAILS DENOTE GEAR DOWN
LONGITUDINAL CONTROL SENSITIVITY
DEG/SEC/SEC
INCH STICK DISPLACEMENT
0
10
20
0
20
40
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 21
LONGITUDINAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
15,600	299.8	-440	247	11.7	HOVER (IGE-15 FT)	CLEAN (GEAR DN)

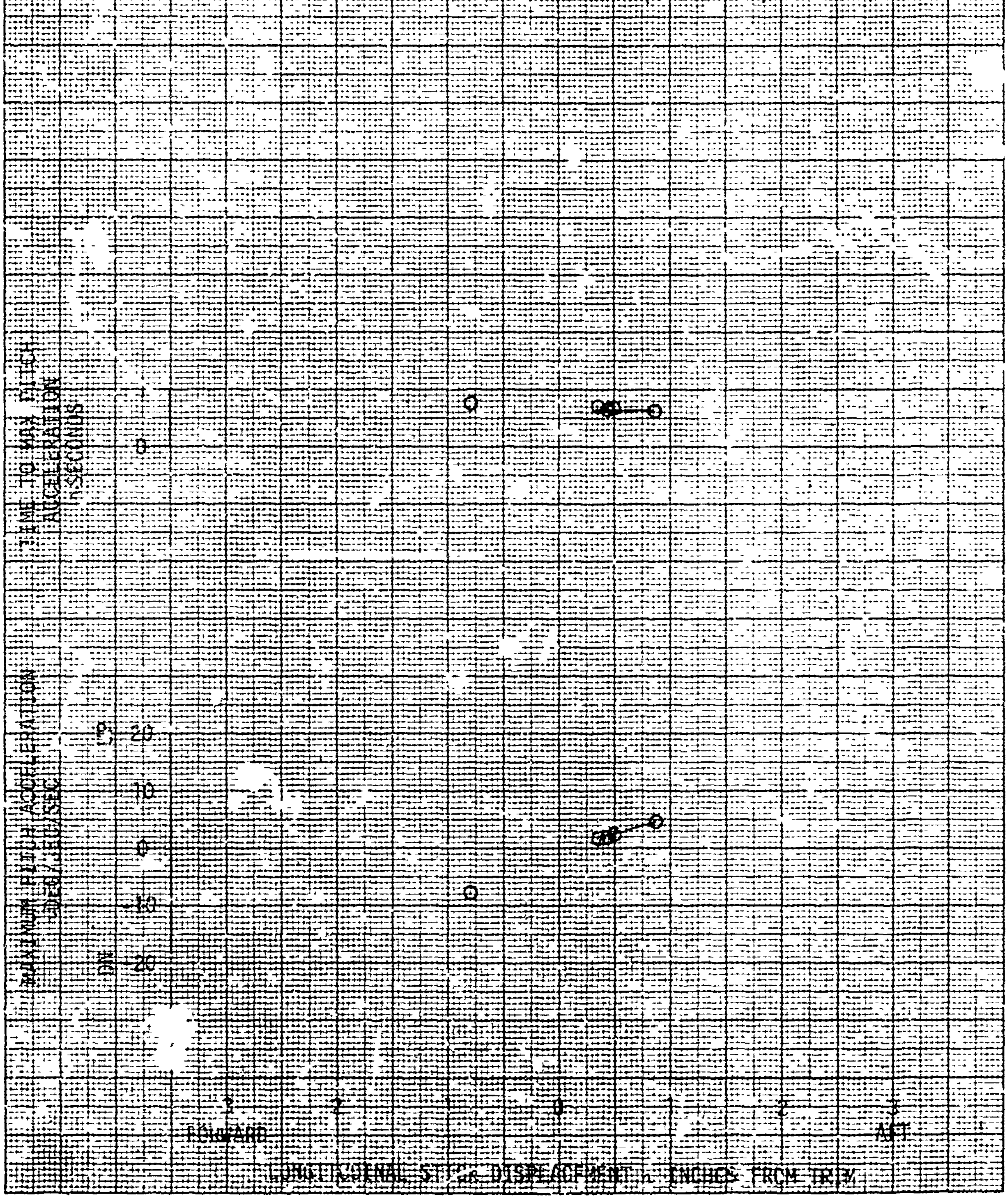


FIGURE 22
LONGITUDINAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CAL'B AIRSPEED ~KT	CONFIGURATION
18,380	298.9	3350	247	8.3	63	CLEAN (GEAR DN)

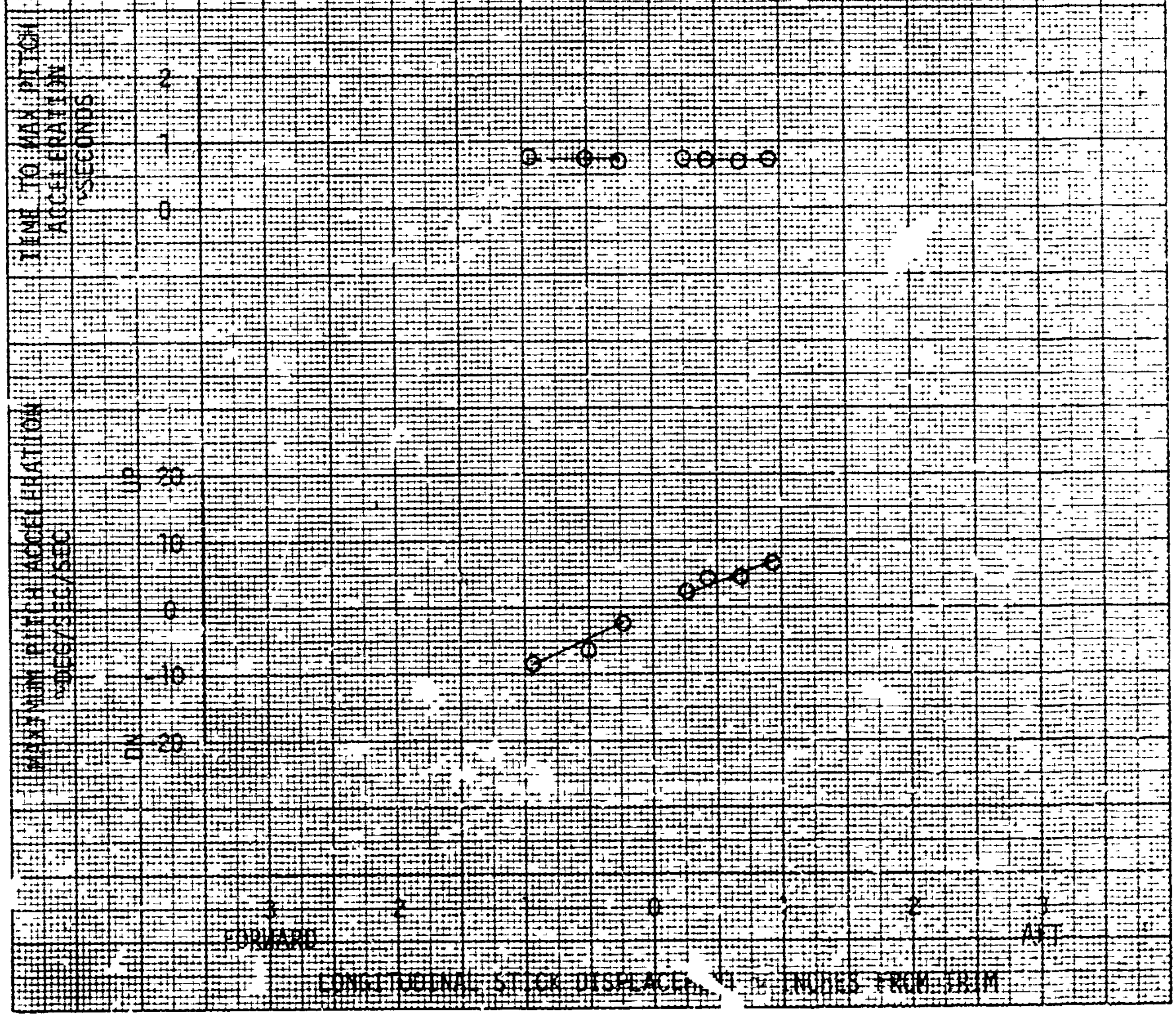


FIGURE 23
LONGITUDINAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CAL. AIRSPEED ~KT	CONFIGURATION
18,510	299.8	4140	246	7.0	101	CLEAN (GEAR UP)

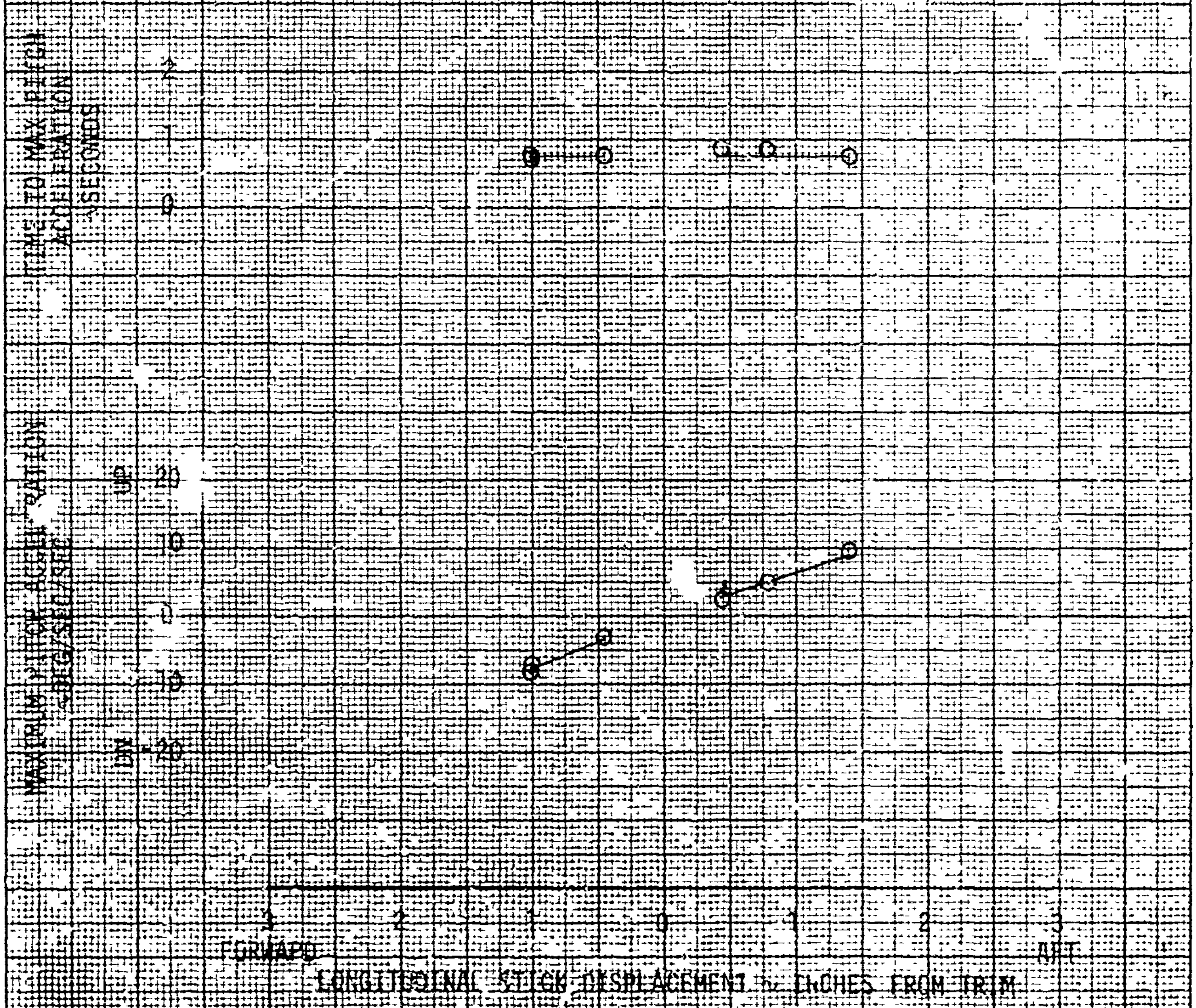


FIGURE 24
LONGITUDINAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,300	299.7	4600	248	5.4	149	CLEAN (GEAR UP)

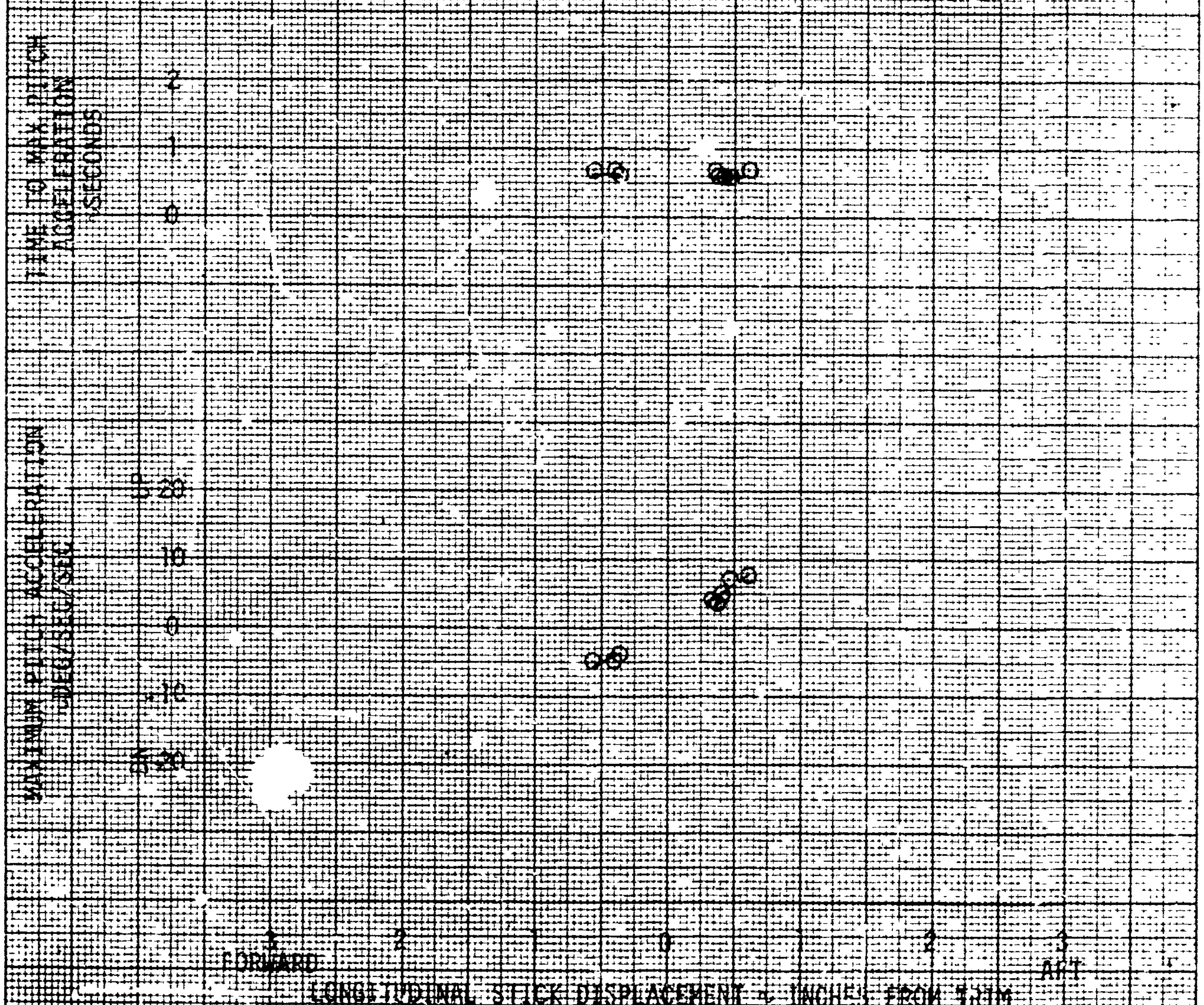


FIGURE 25
SUMMARY LATERAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
18,330	299.7	-450	247	CLEAN	HOVER (IGE-15 FT)
18,070	299.5	4220	247	CLEAN	LEVEL FLIGHT
18,360	299.7	4600	247	CLEAN	POWERED DESCENT

NOTES:
1. POINTS DERIVED FROM FIGURES 26 THROUGH 31 AT ONE INCH
2. TAILS DENOTE GEAR DOWN
3. SOLID SYMBOLS DENOTE POWERED DESCENT

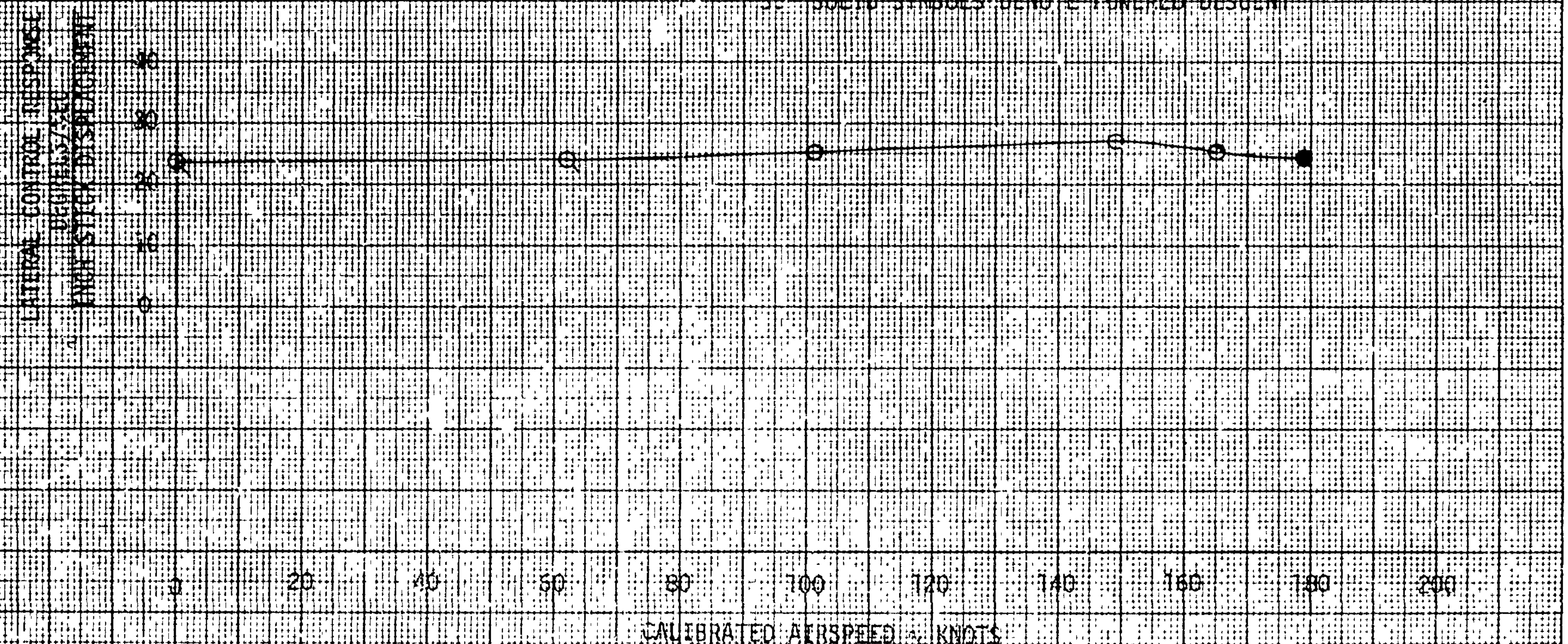


FIGURE 26
LATERAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,330	209.7	-450	247	11.8	HOVER (IGE~15 FT)	CLEAN (GEAR DN)

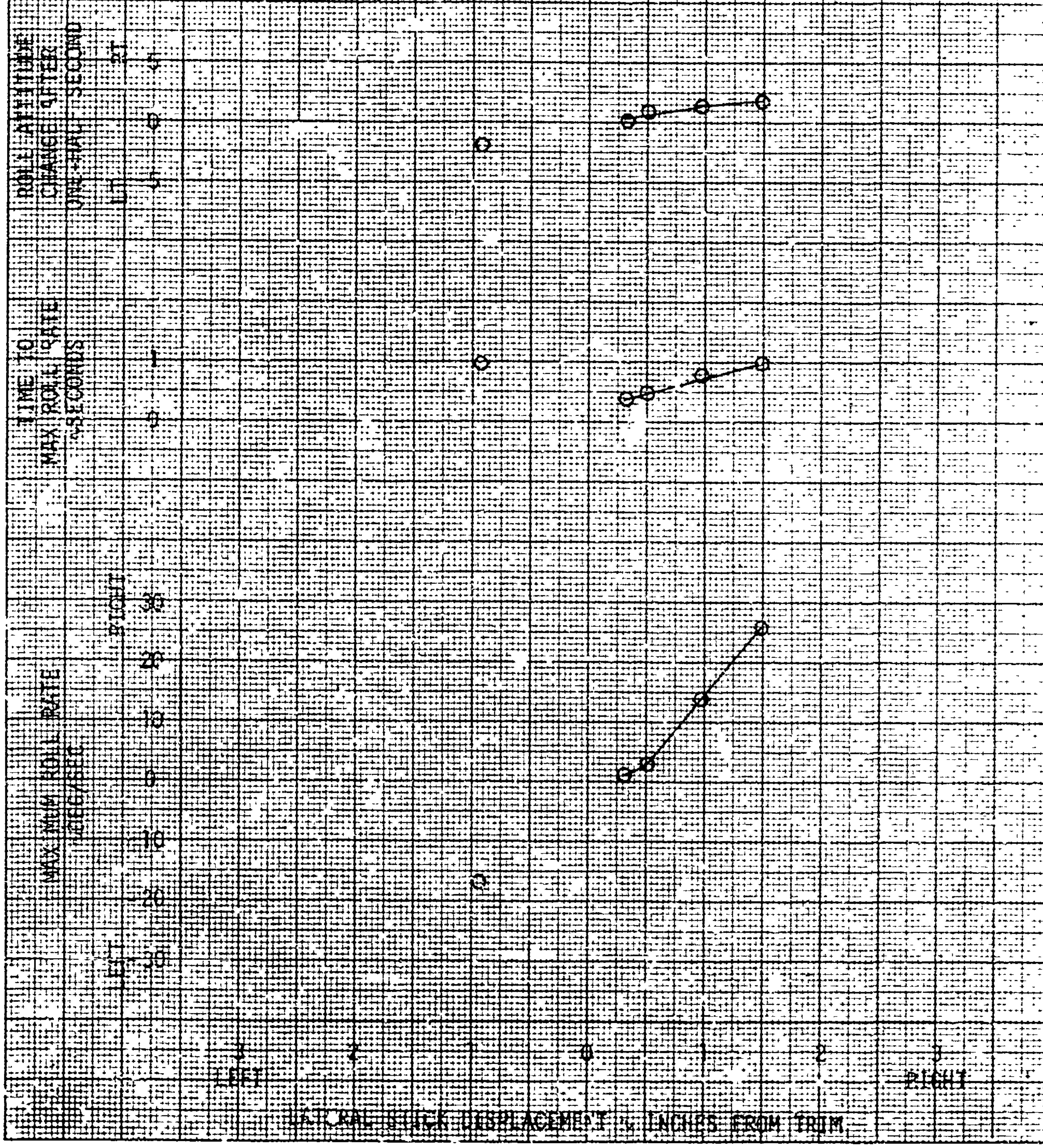
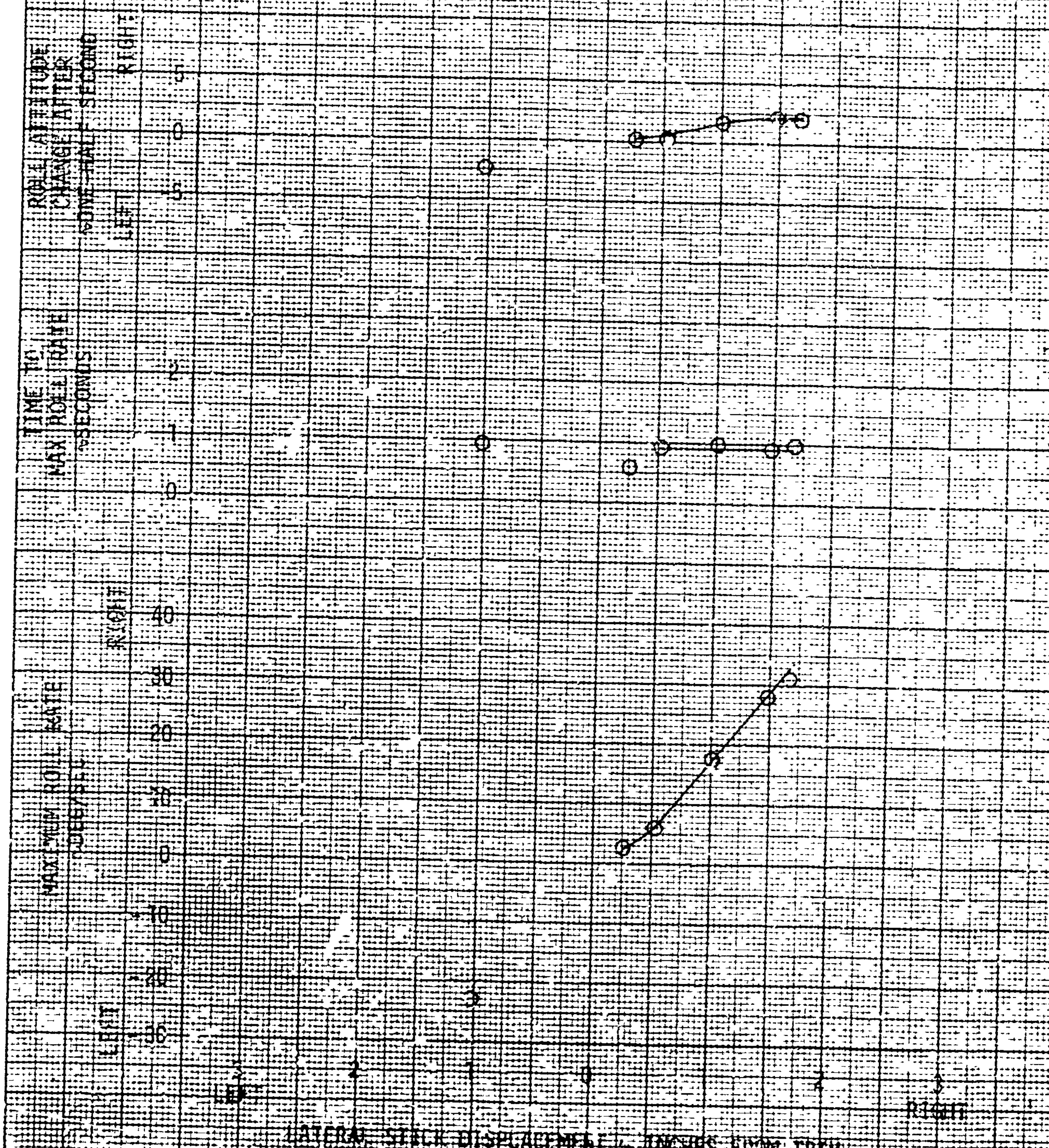


FIGURE 27
LATERAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,160	298.8	3900	247	8.2	62	CLEAN (GEAR DN)



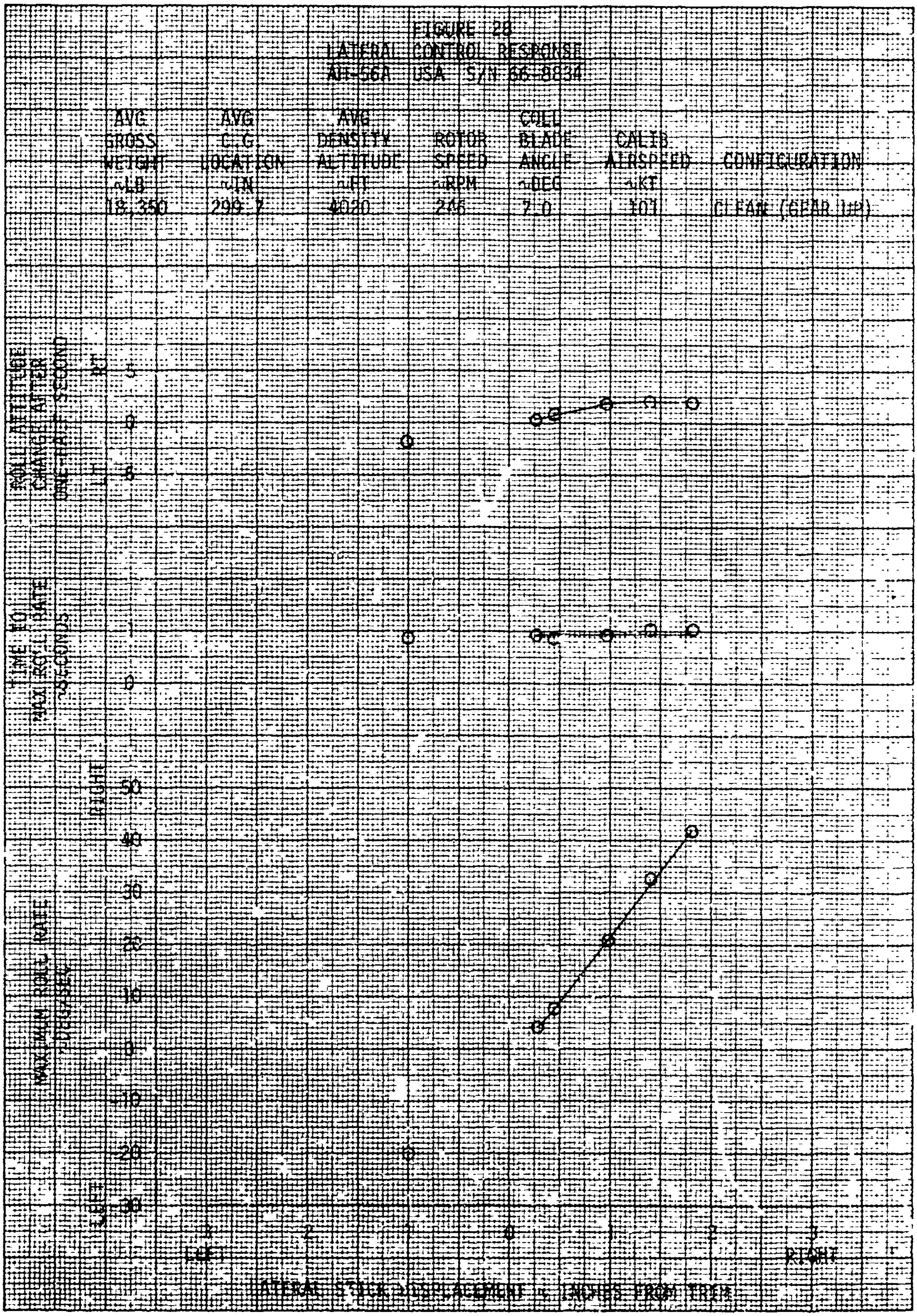


FIGURE 28
LATERAL CONTROL RESPONSE
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 18,350
AVG C.G. LOCATION ~IN 299.7
AVG DENSITY ALTITUDE ~FT 4020
ROTOR SPEED ~RPM 246
COLL BLADE ANGLE ~DEG 7.0
CALIB AIRSPEED ~KT 101
CONFIGURATION CLEAN (GEAR UP)
ROLL ATTITUDE CHANGE AFTER ONE-HALF SECOND
RT
LT
TIME TO MAX ROLL RATE ~SECONDS
MAXIMUM ROLL RATE ~DEG/SEC
RIGHT
LEFT
LEFT
RIGHT
LATERAL STICK DISPLACEMENT ~ INCHES FROM TRIM

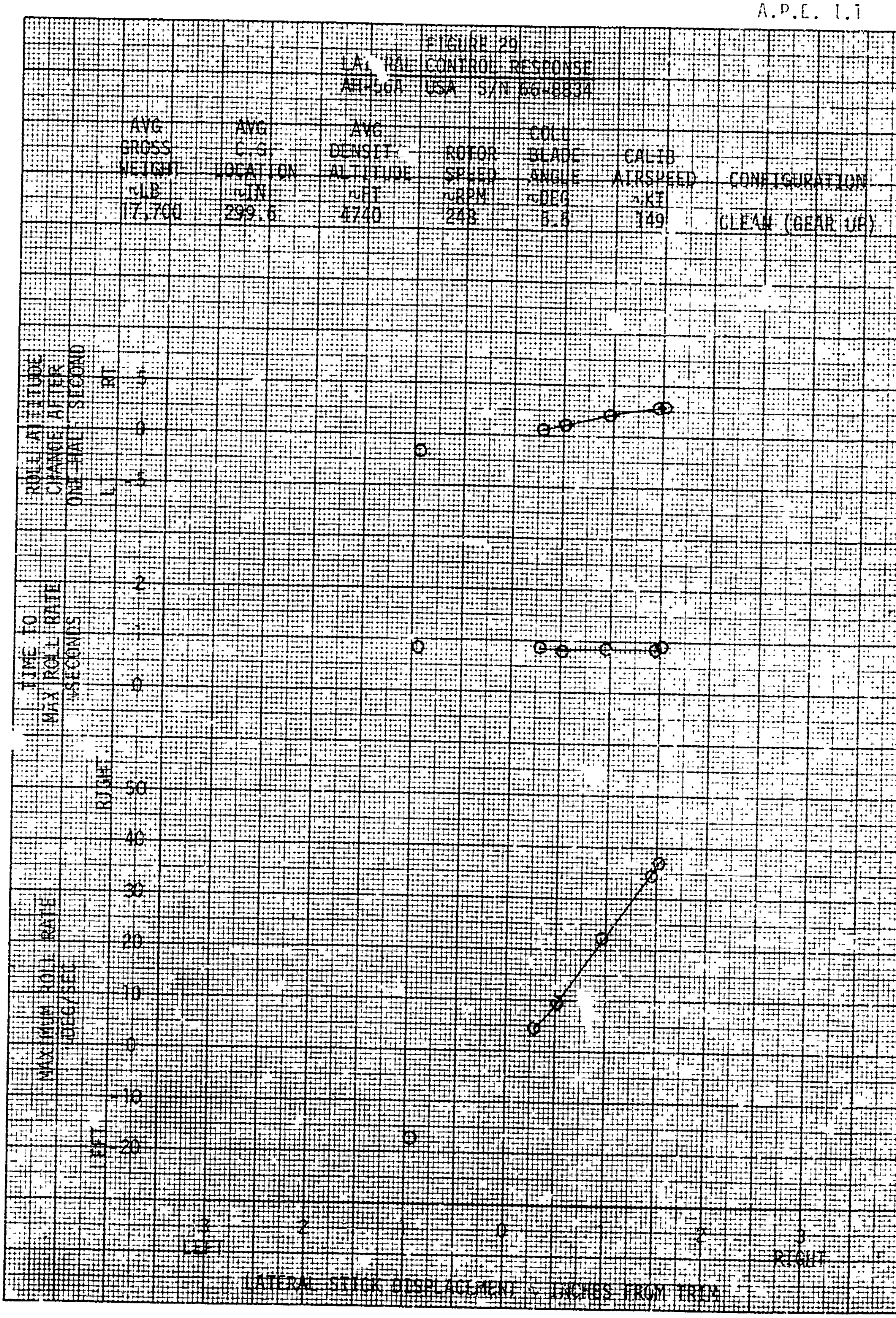


FIGURE 29
LATERAL CONTROL RESPONSE
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 17,700
AVG C.G. LOCATION ~IN 299.6
AVG DENSITY ALTITUDE ~FT 4740
ROTOR SPEED ~RPM 248
COLL BLADE ANGLE ~DEG 6.8
CALIB AIRSPEED ~KT 149
CONFIGURATION CLEAN (GEAR UP)
ROLL ATTITUDE CHANGE AFTER ONE HALF SECOND
RT
L
TIME TO MAX ROLL RATE ~SECONDS
MAXIMUM ROLL RATE DEG/SEC
RIGHT
LEFT
50
40
30
20
10
0
-10
-20
3
2
1
0
1
2
3
LEFT
RIGHT
LATERAL STICK DISPLACEMENT ~ INCHES FROM TRIM

FIGURE 30
LATERAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,290	299.7	5340	248	5.1	165	CLEAN (GEAR UP)

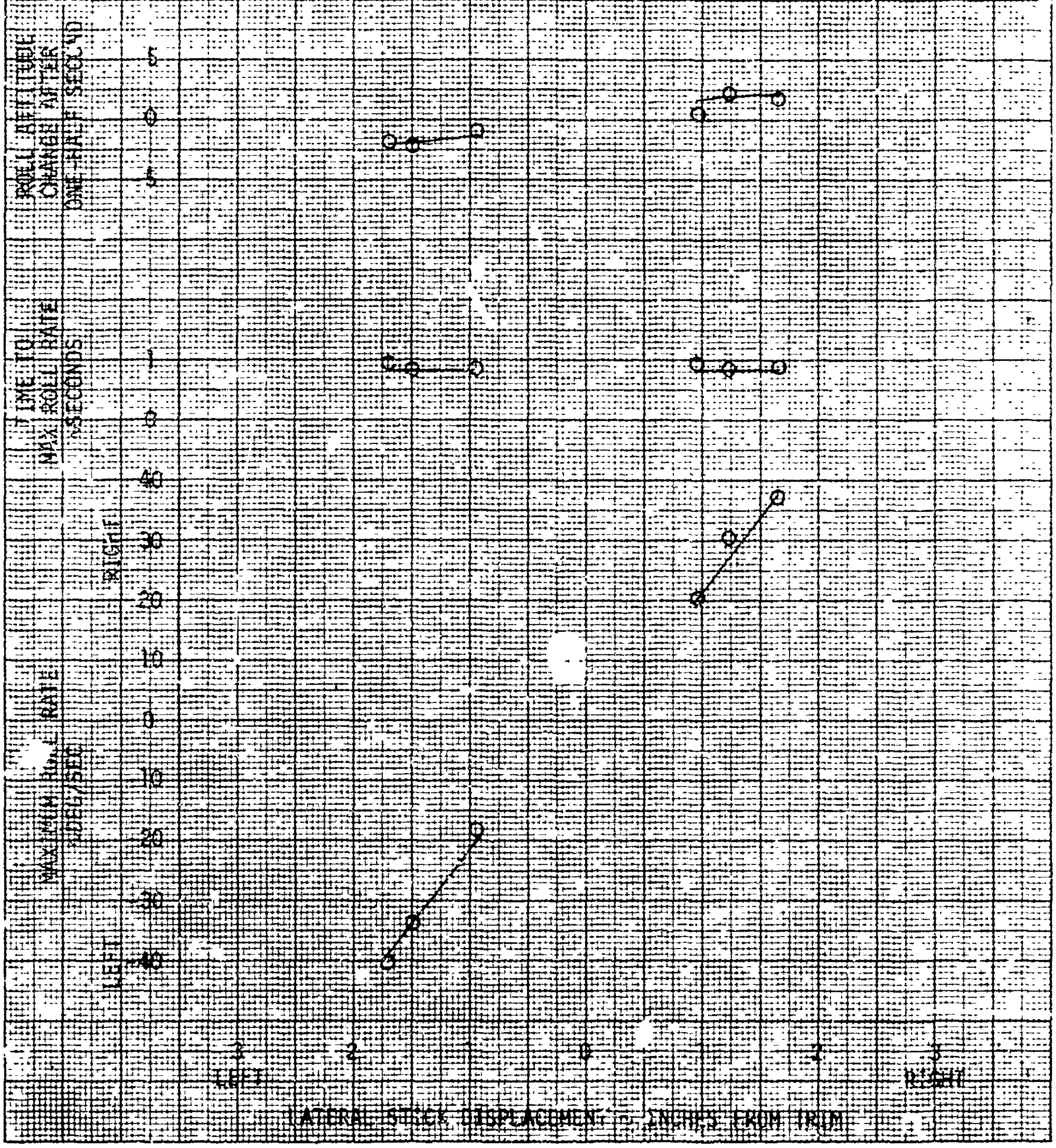


FIGURE 31
LATERAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,300	299.7	3370	248	5.1	179	CLEAN (GEAR UP)

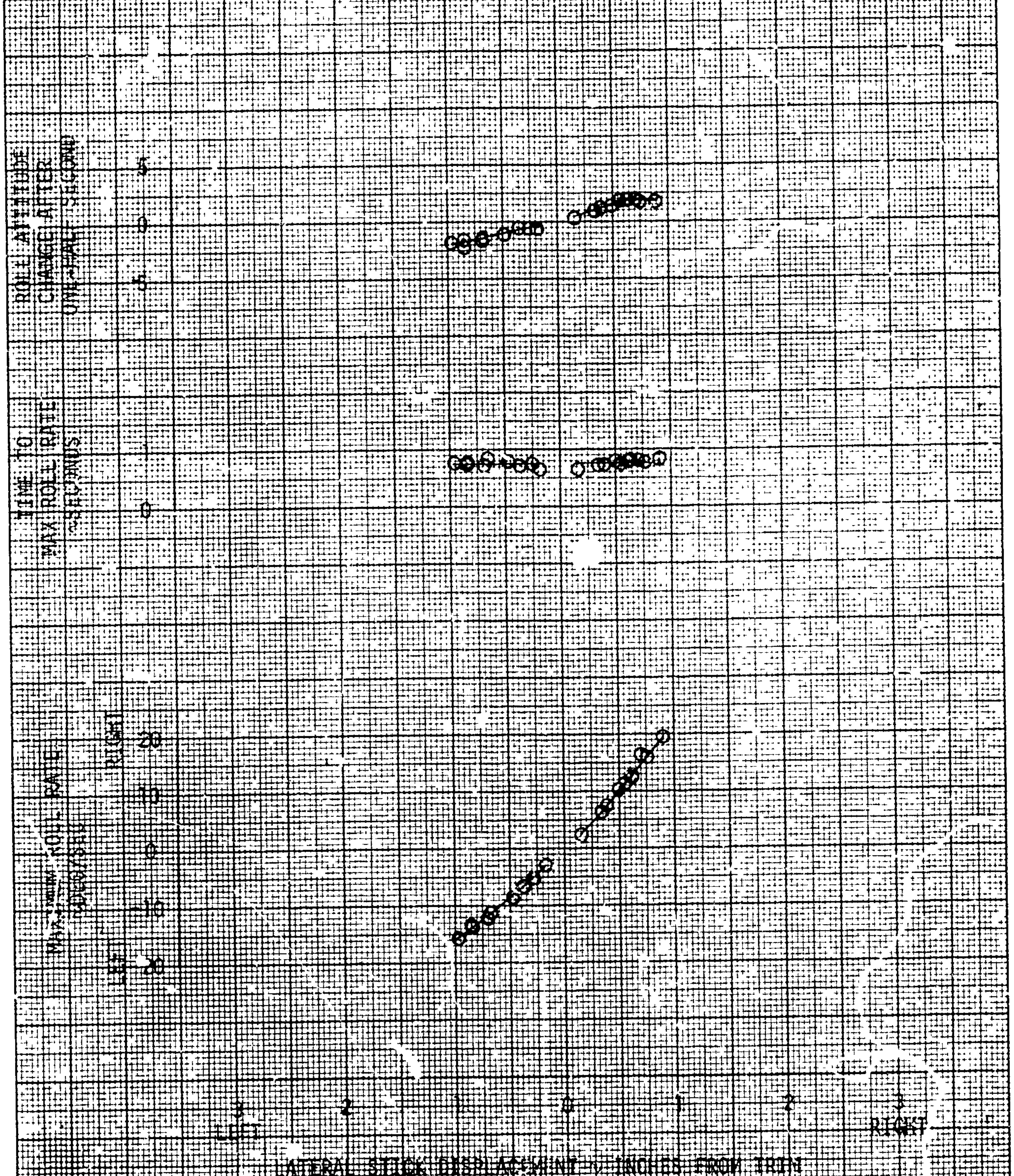
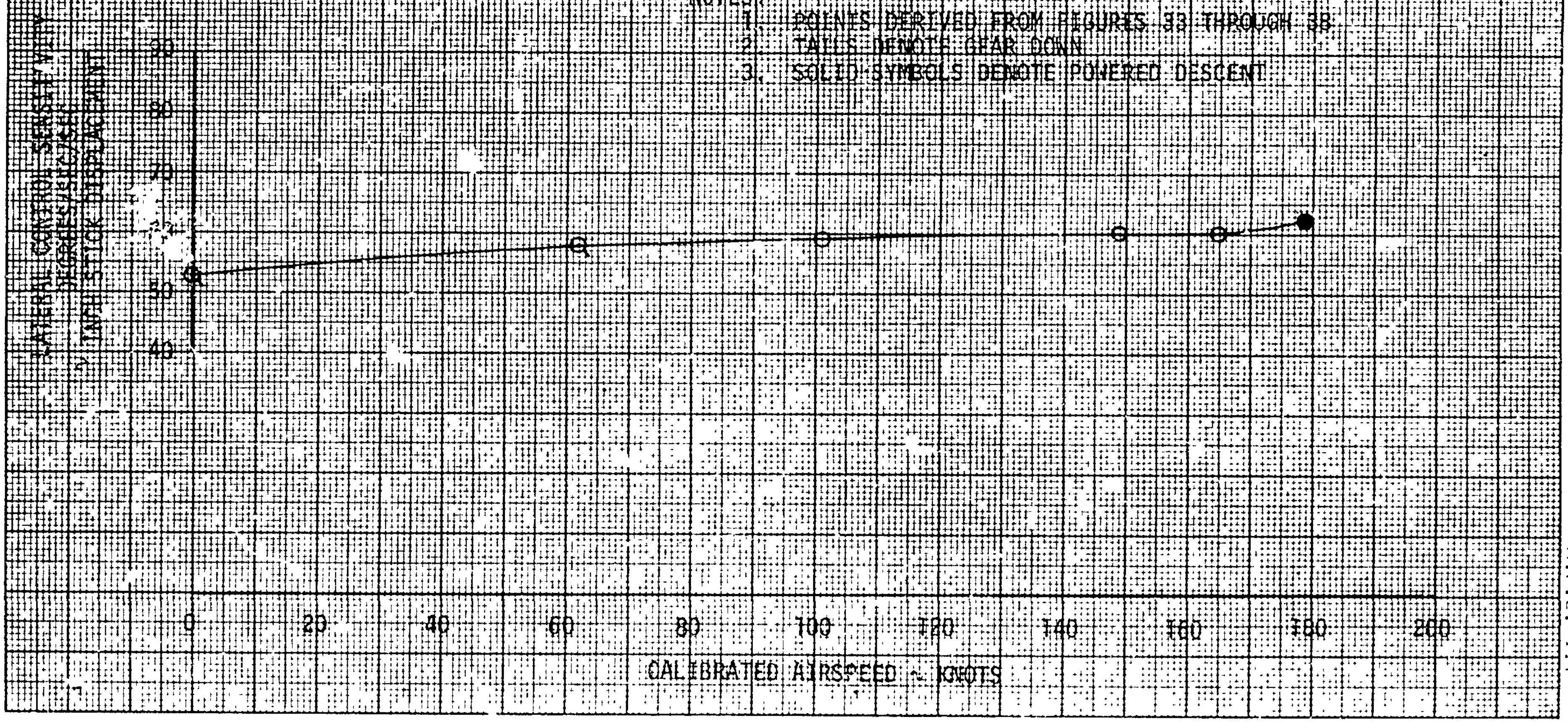


FIGURE 32
SUMMARY LATERAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
18,330	299.7	-450	247	CLEAN	HOVER(IGE-5 FT)
18,070	299.3	4220	247	CLEAN	LEVEL FLIGHT
18,300	299.7	4600	247	CLEAN	POWERED DESCENT

NOTES:
1. POINTS DERIVED FROM FIGURES 33 THROUGH 38.
2. TAILS DENOTE GEAR DOWN
3. SOLID SYMBOLS DENOTE POWERED DESCENT



A.P.E. I.1

FIGURE 33
LATERAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
8,330	299.7	-450	247	11.8	HOVER (IGE 16 FT)	CLEAN (GEAR DN)

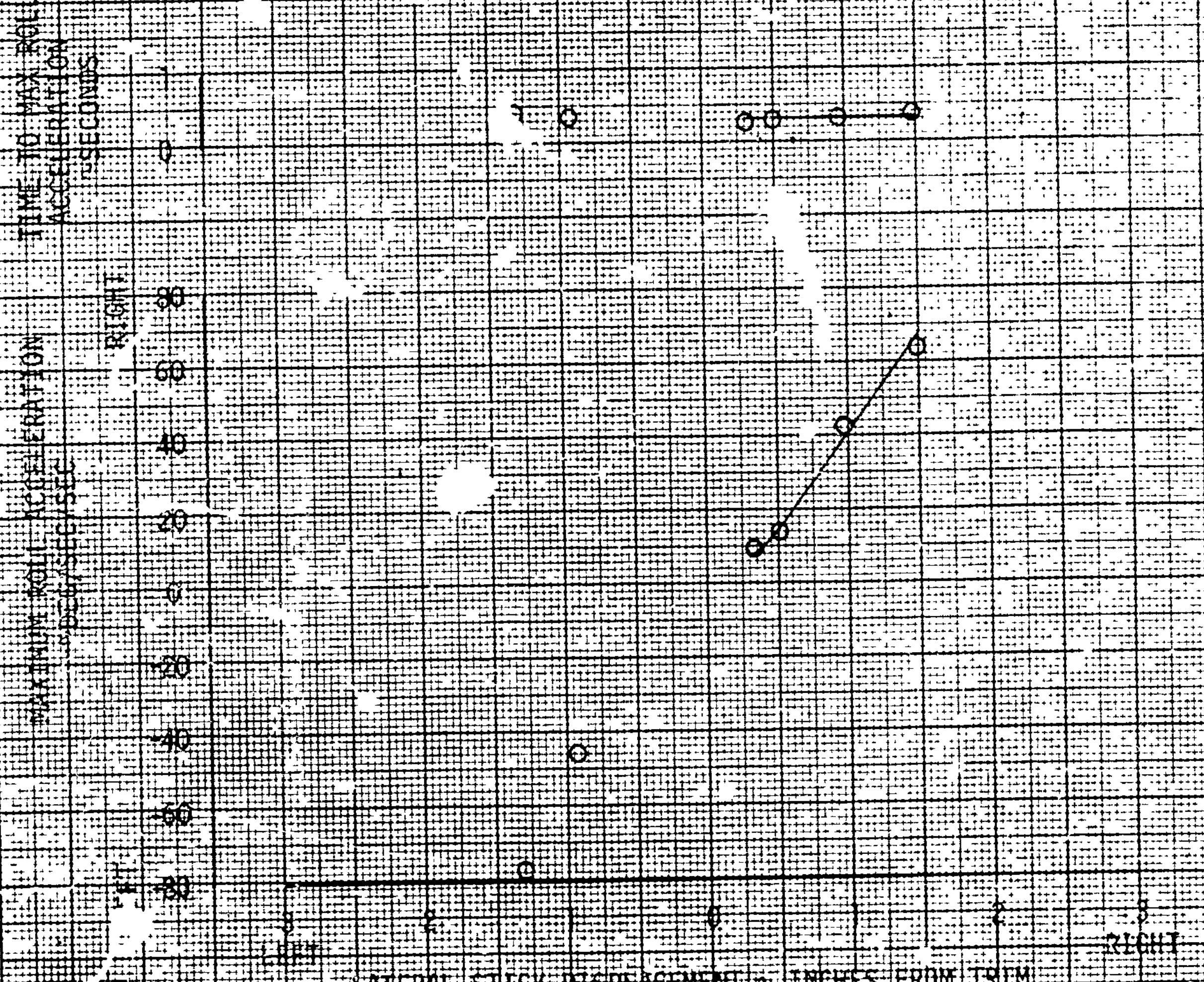


FIGURE 34

LATERAL CONTROL SENSITIVITY

AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,160	298.8	3900	[illegible]	[illegible]	62	CLEAN (GEAR ON)

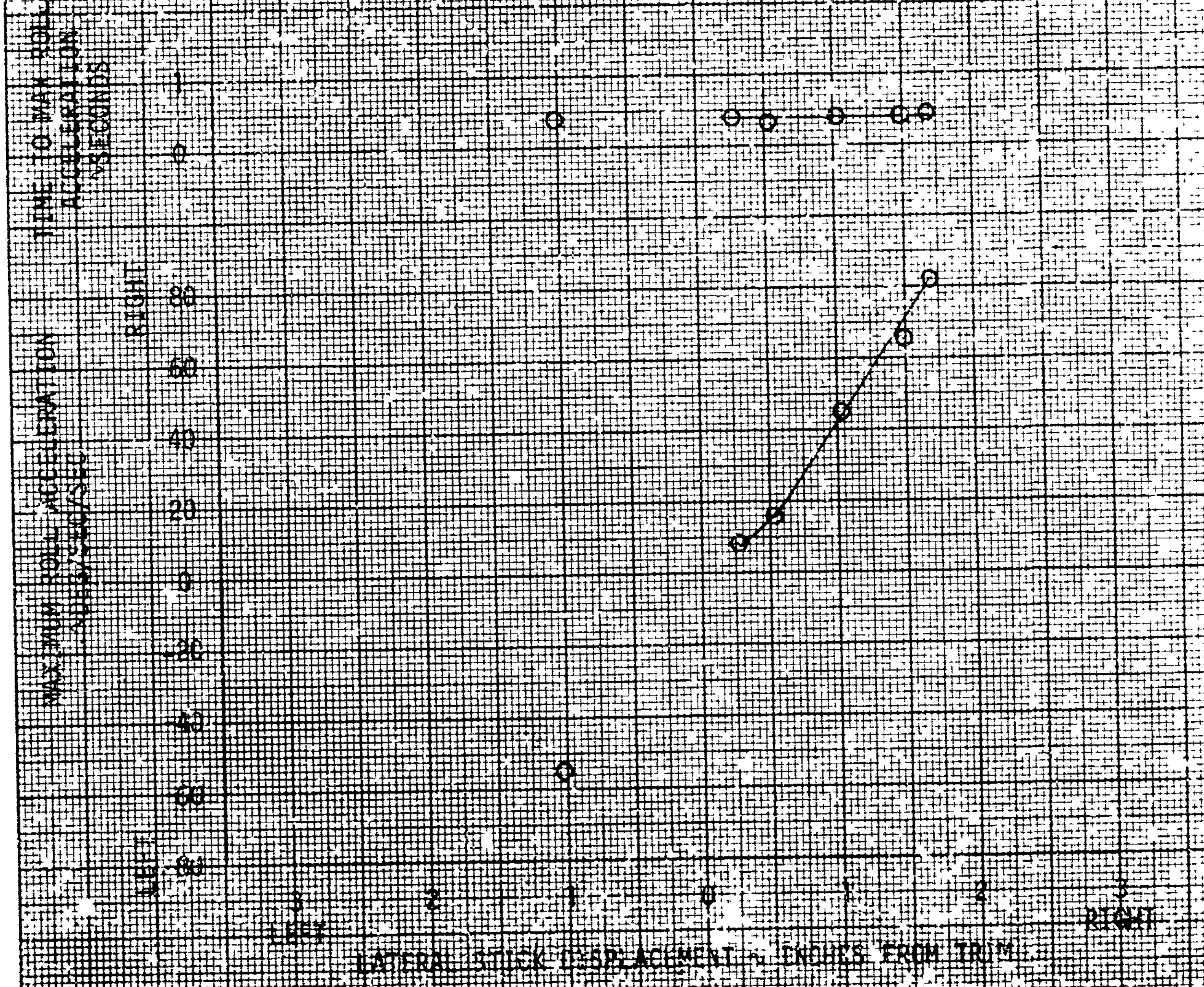


FIGURE 35
LATERAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,35	299.7	4020	296	7.0	101	[illegible] (GEAR UP)

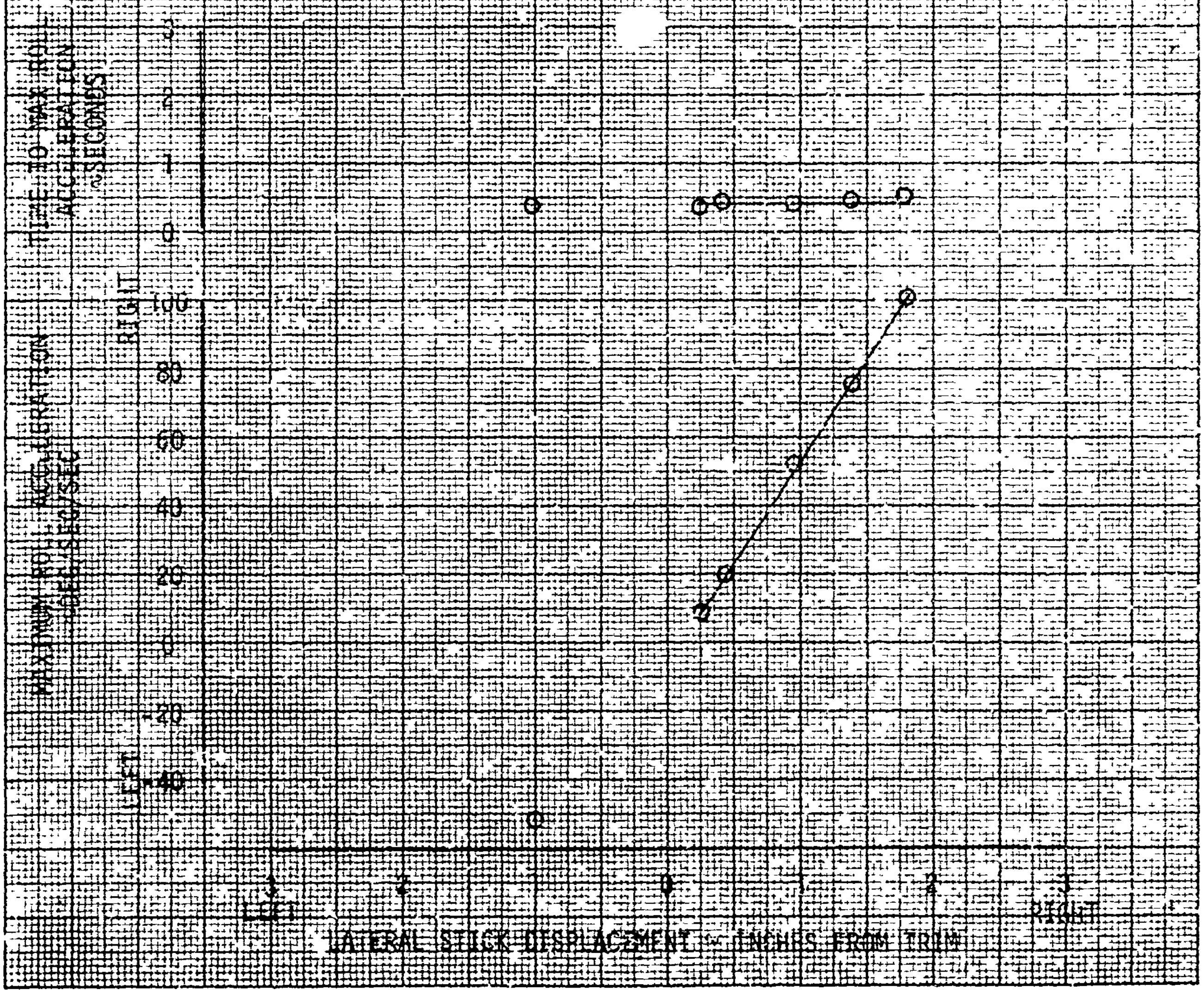


FIGURE 26
LATERAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
17,700	299.5	4740	248	5.5	149	CLEAN (GEAR UP)

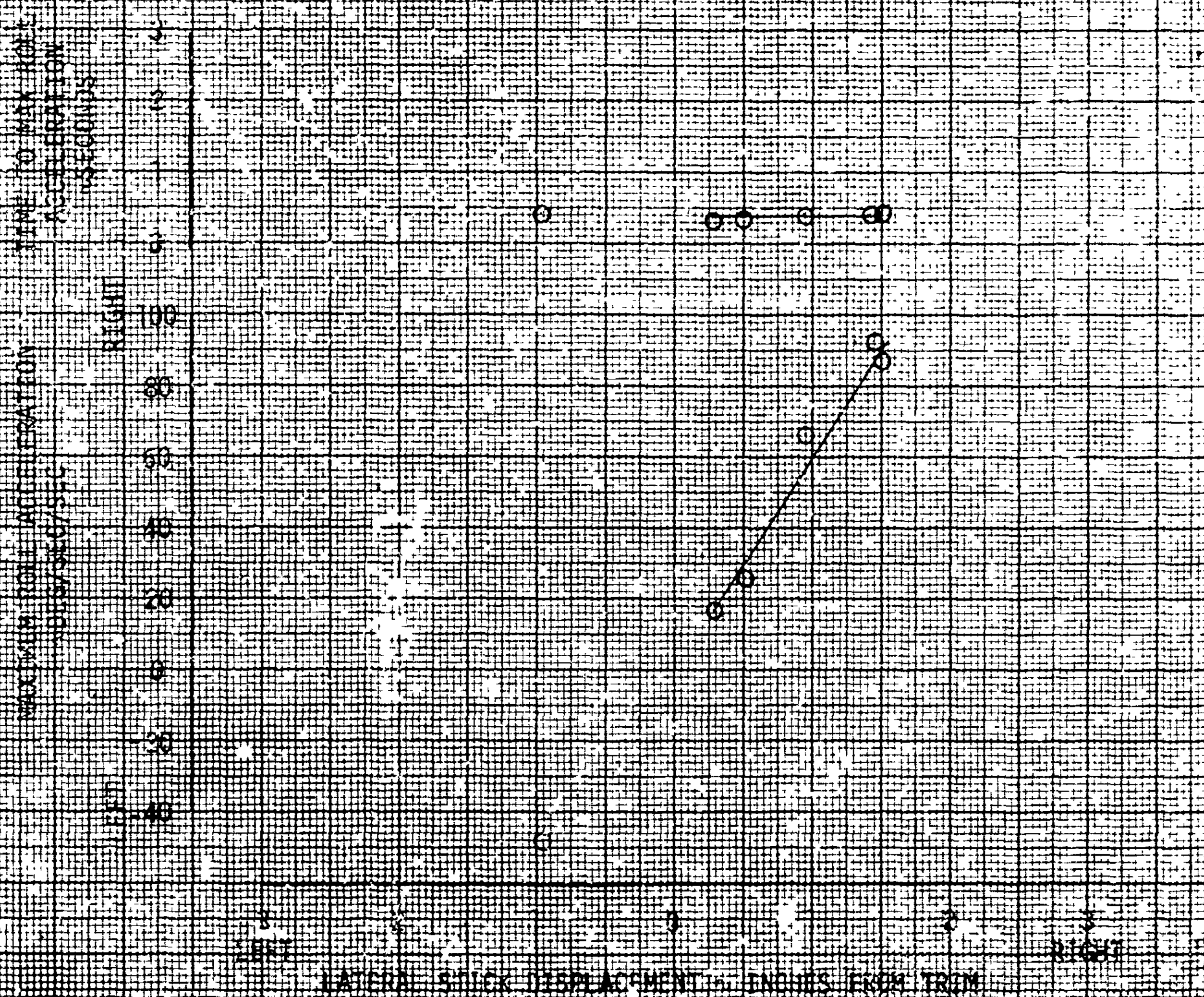


FIGURE 37
LATERAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,290	299.7	5840	248	5.1	165	CLEAN (GEAR UP)

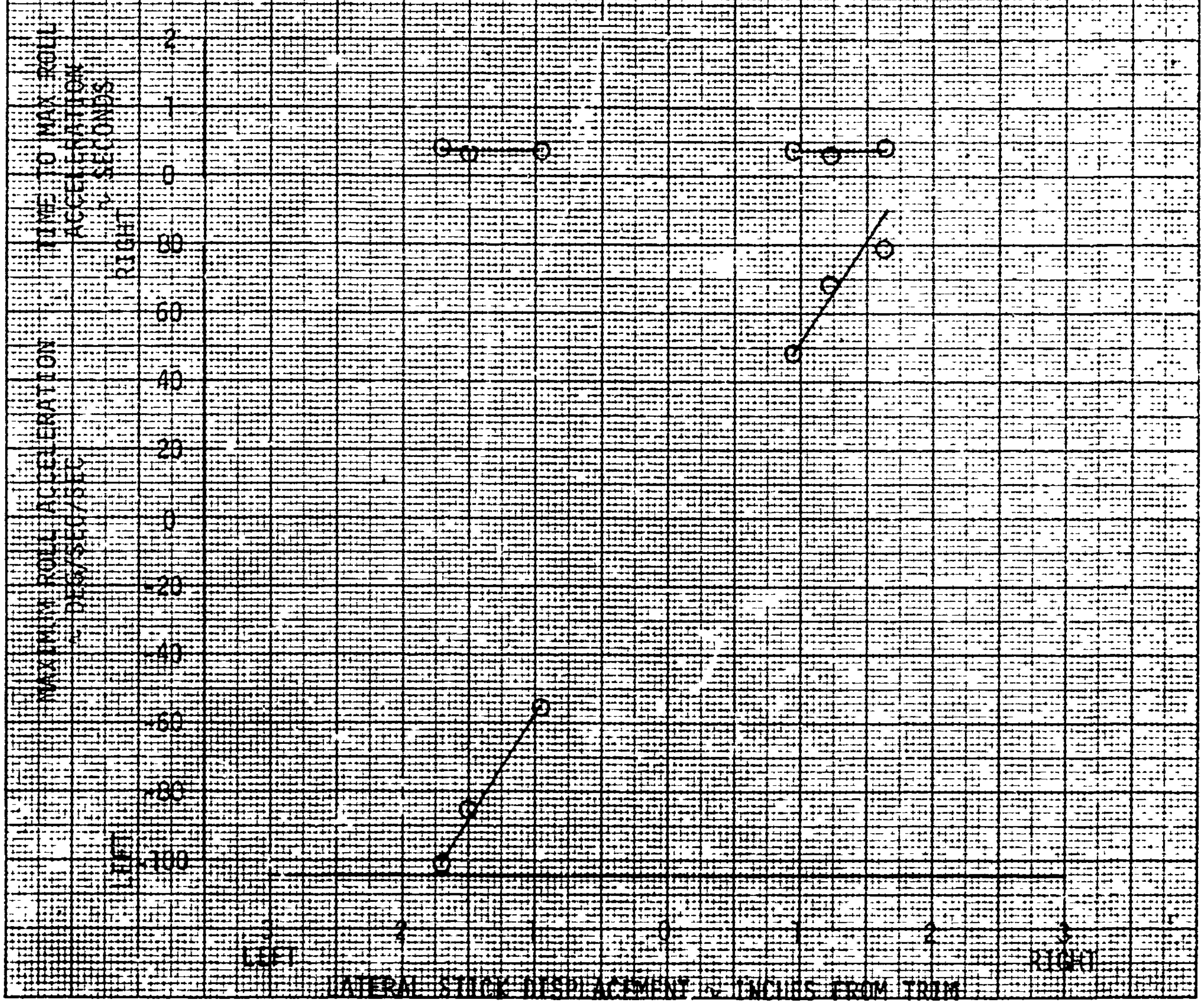


FIGURE 38
LATERAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,300	299.7	3370	248	5.1	179	CLEAN (GEAR UP)

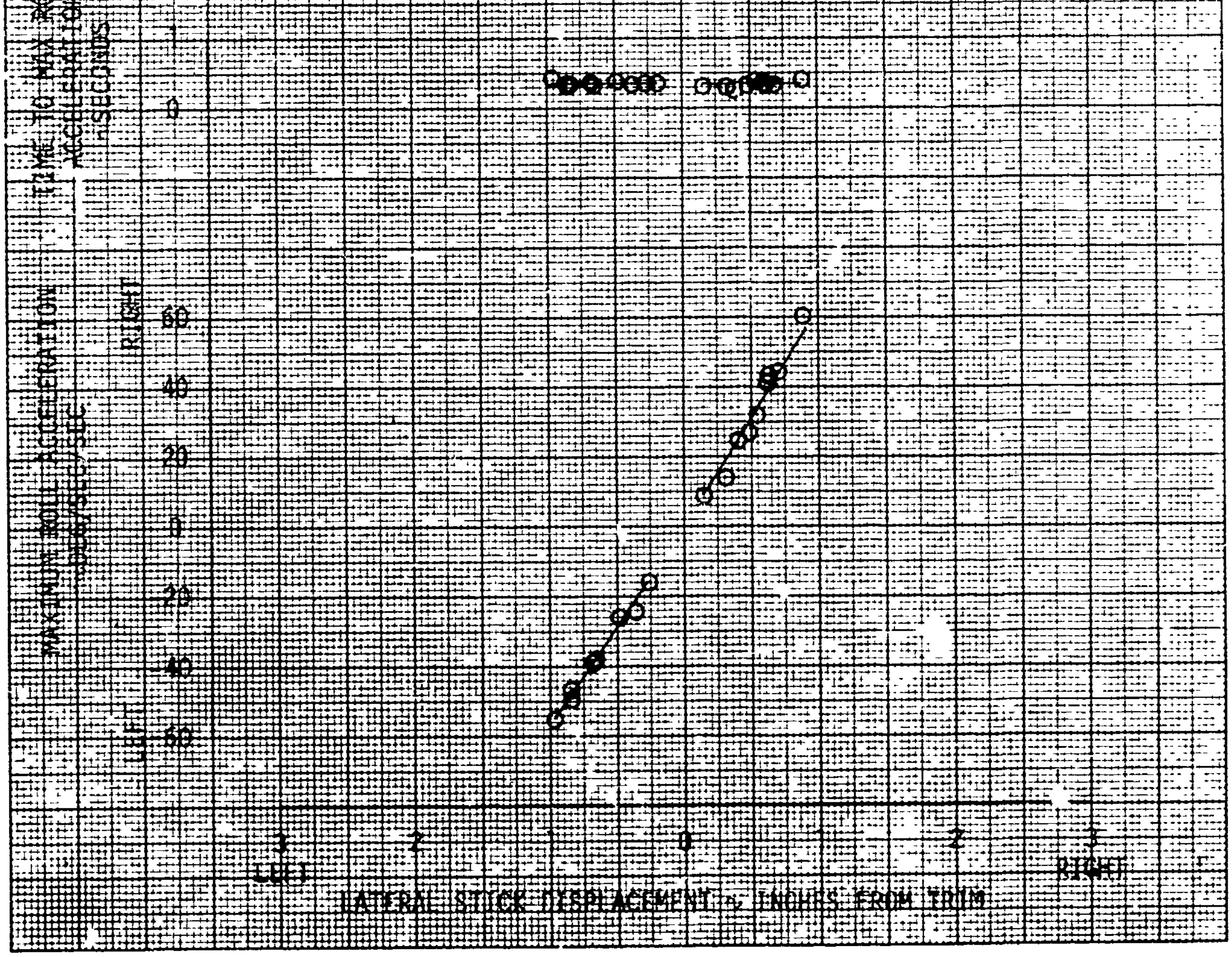


FIGURE 39
SUMMARY DIRECTIONAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
17,920	229.6	1590	248	CLEAN	HOVER (IGE~5 FT)
18,150	229.3	4710	248	CLEAN	LEVEL FLIGHT

NOTES:
1. POINTS DERIVED FROM FIGURES 41 THROUGH 42
   TAILS DENOTE GEAR DOWN
2. MAXIMUM RATE NOT OBTAINED IN A HOVER.

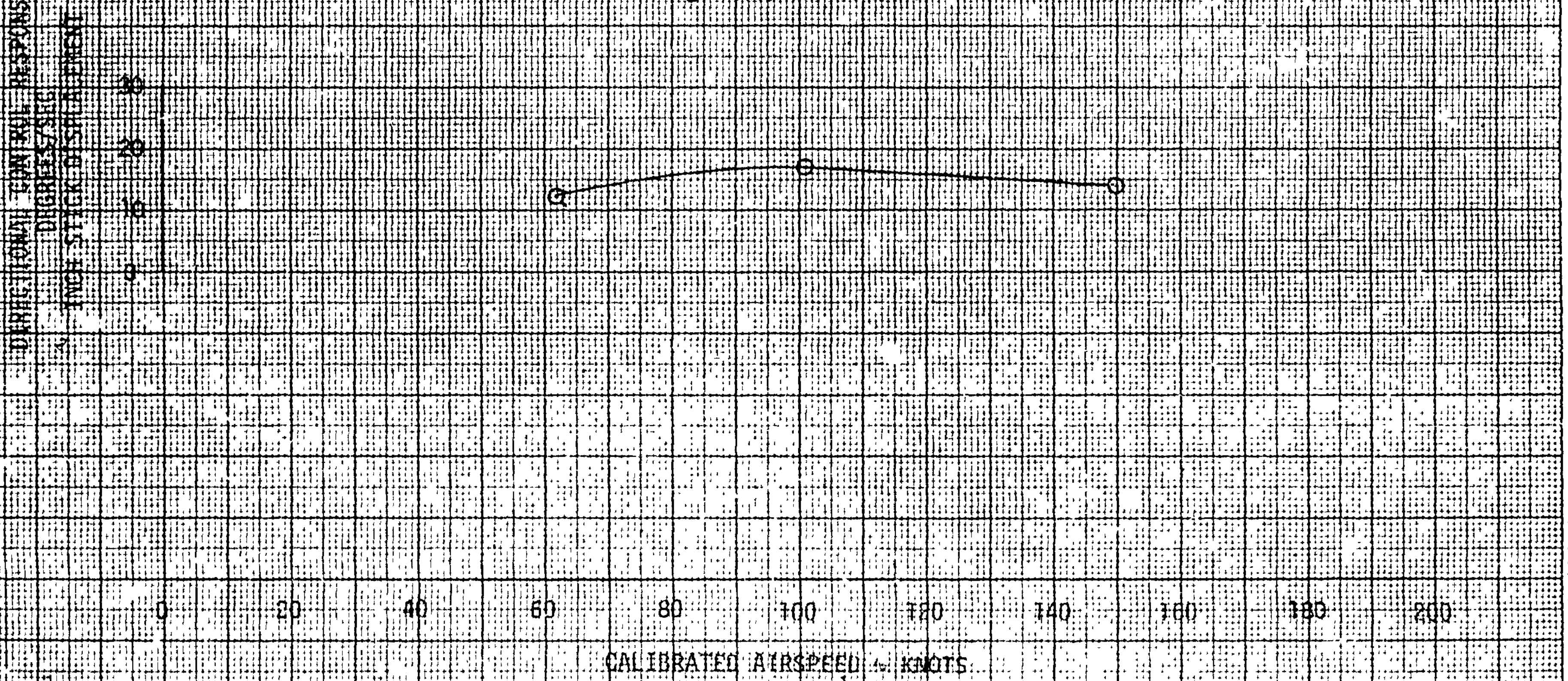


FIGURE 40
DIRECTIONAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
17,920	299.6	-590	248	11.5	HOVER (IGE~15 FT)	CLEAN (GEAR DN)

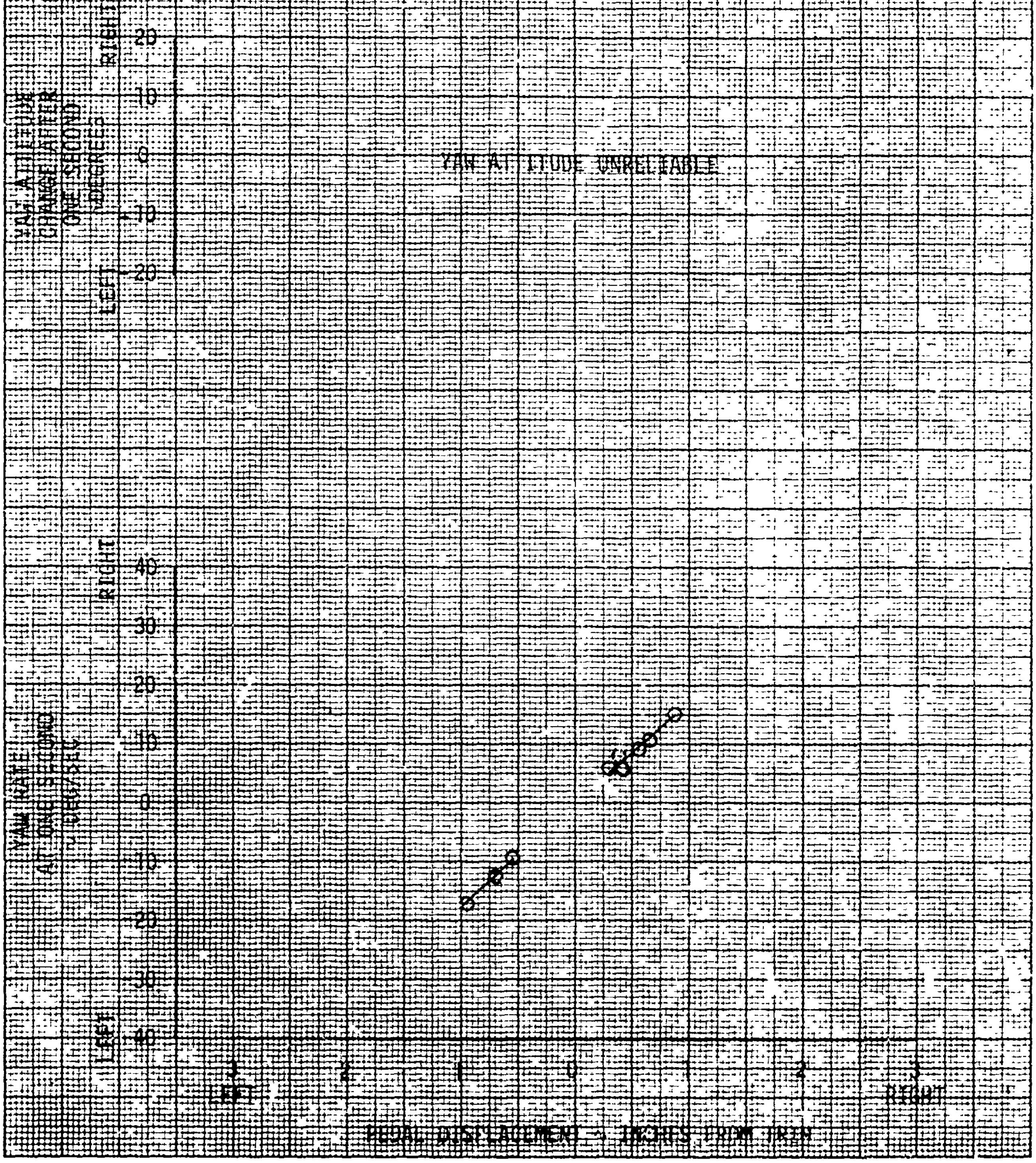
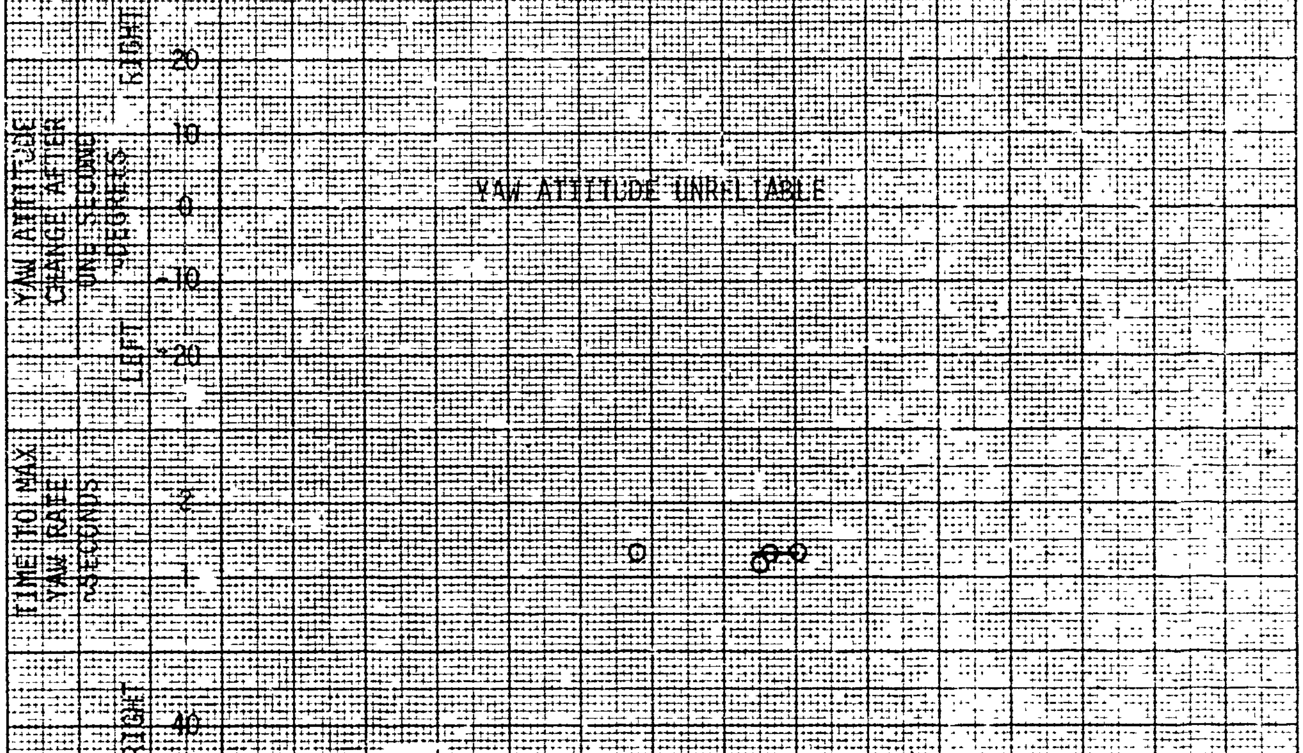


FIGURE 41
DIRECTIONAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,240	298.7	4320	248	8.3	62	CLEAN (GEAR ON)



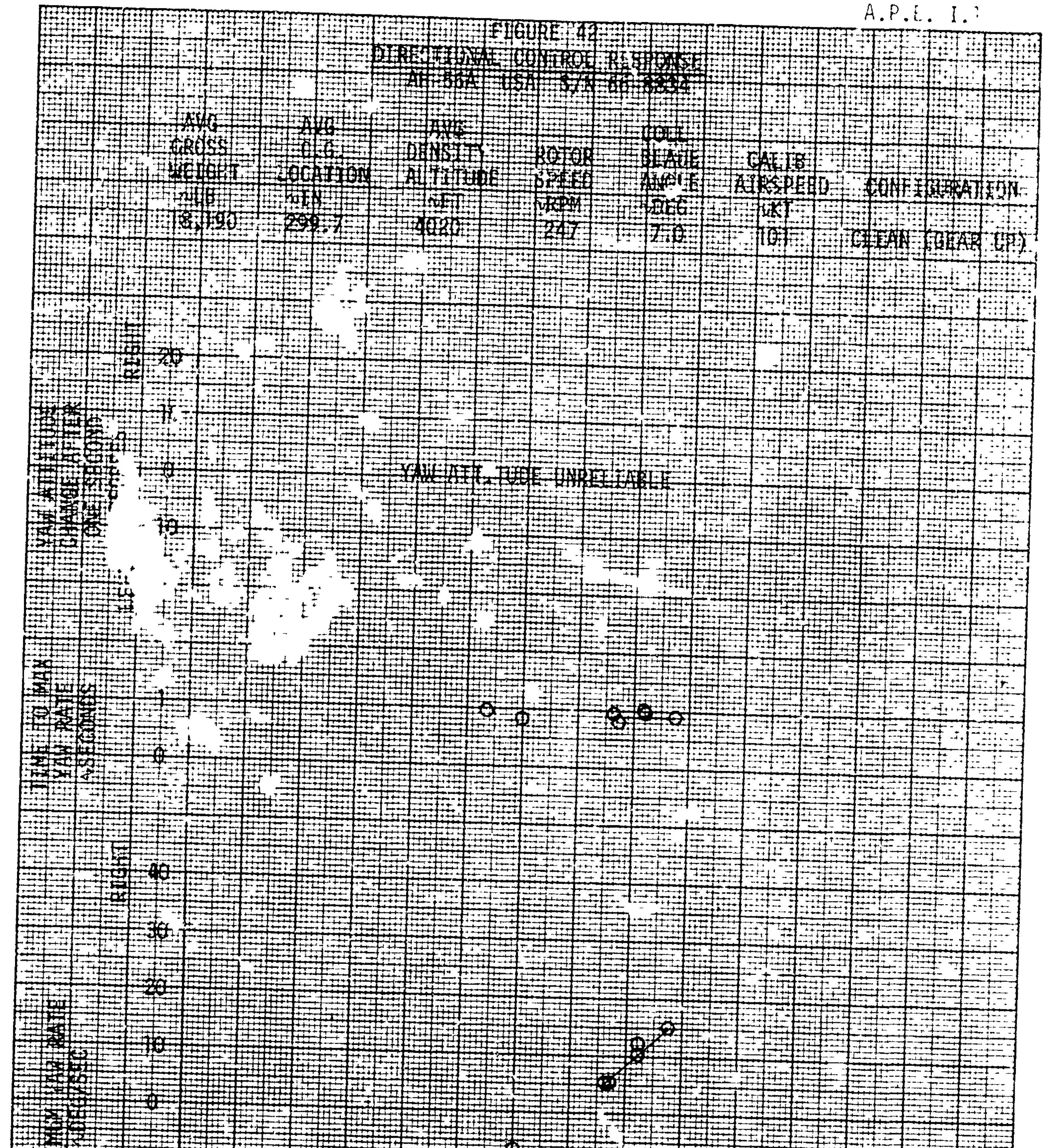

A.P.E. I.?
FIGURE 42
DIRECTIONAL CONTROL RESPONSE
AH-56A USA S/N
AVG GROSS WEIGHT ~LB
18,190
AVG C.G. LOCATION ~IN
299.7
AVG DENSITY ALTITUDE ~FT
4020
ROTOR SPEED ~RPM
247
COLL BLADE ANGLE ~DEG
7.0
CALIB AIRSPEED ~KT
101
CONFIGURATION
CLEAN (GEAR UP)
YAW ATTITUDE CHANGE AFTER ONE SECOND
RIGHT
LEFT
YAW ATTITUDE UNRELIABLE
TIME TO MAX YAW RATE ~SECONDS
MAXIMUM YAW RATE ~DEG/SEC
RIGHT
LEFT
LEFT
RIGHT
PEDAL DISPLACEMENT ~ INCHES FROM TRIM

FIGURE 43
DIRECTIONAL CONTROL RESPONSE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,010	299.6	5800	24[illegible]	5.3	150	CLEAN (GEAR UP)

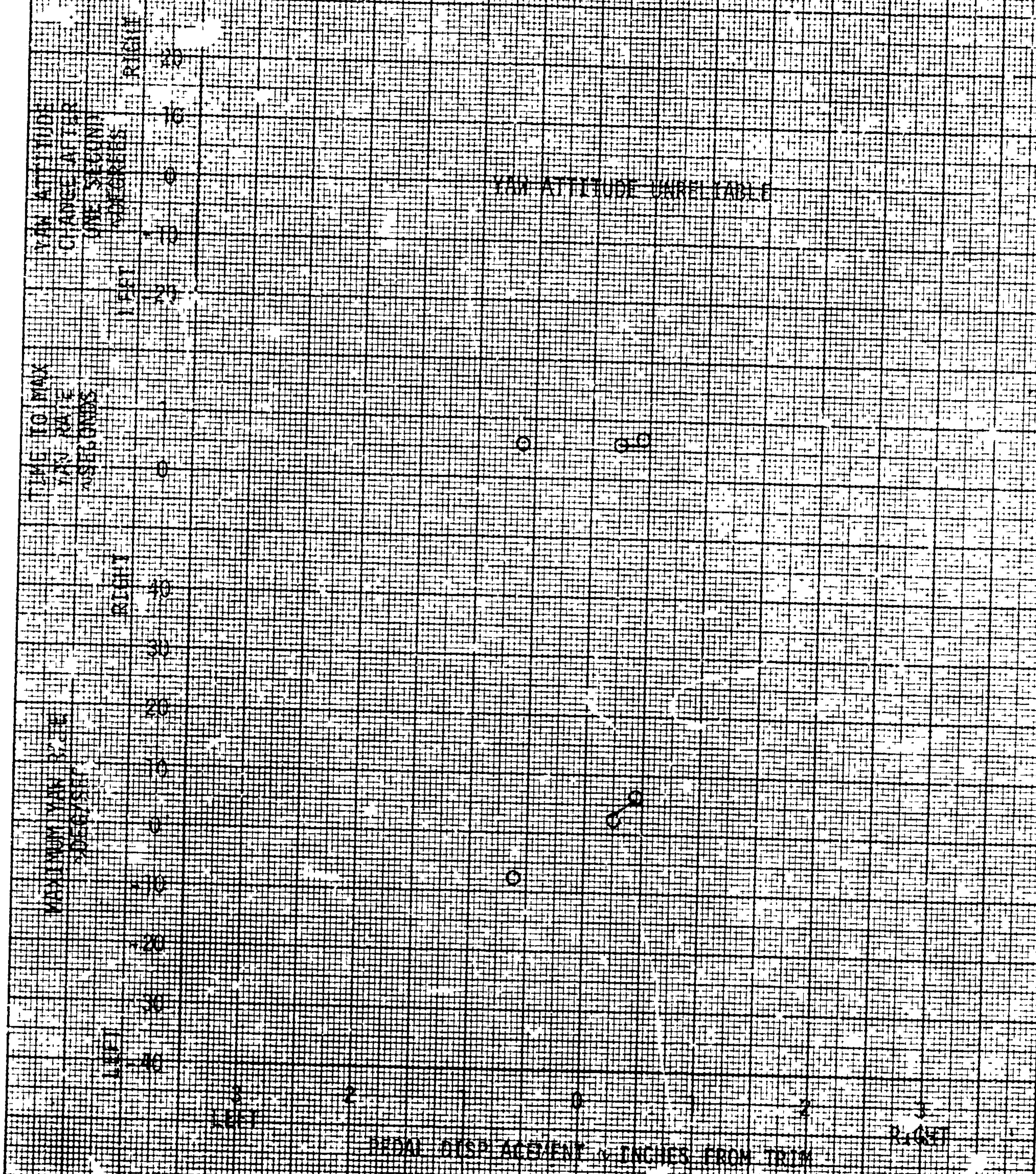
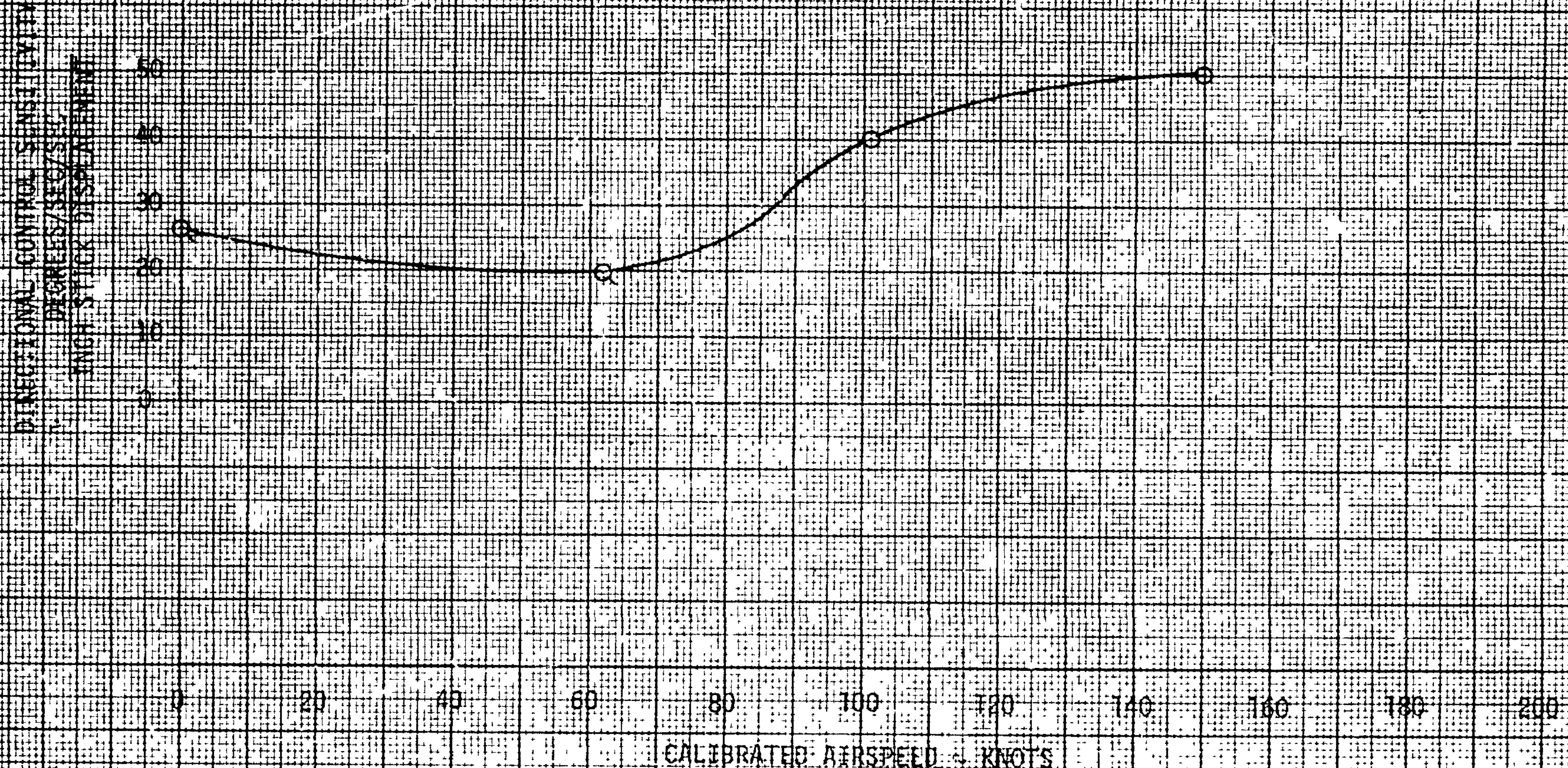


FIGURE 44
SUMMARY DIRECTIONAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
17,290	299.6	596	248	CLEAN	HOVER(IGE-15 FT)
18,150	299.3	4710	248	CLEAN	LEVEL FLIGHT

POINTS DERIVED FROM FIGURES 45 THROUGH 48
TAILS DENOTE GEAR DOWN



A.P.F. 1.1

FIGURE 45
DIRECTIONAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
17,920	299.6	-590	248	11.5	HOVER (IGE 15 FT)	CLEAN (GEAR DN)

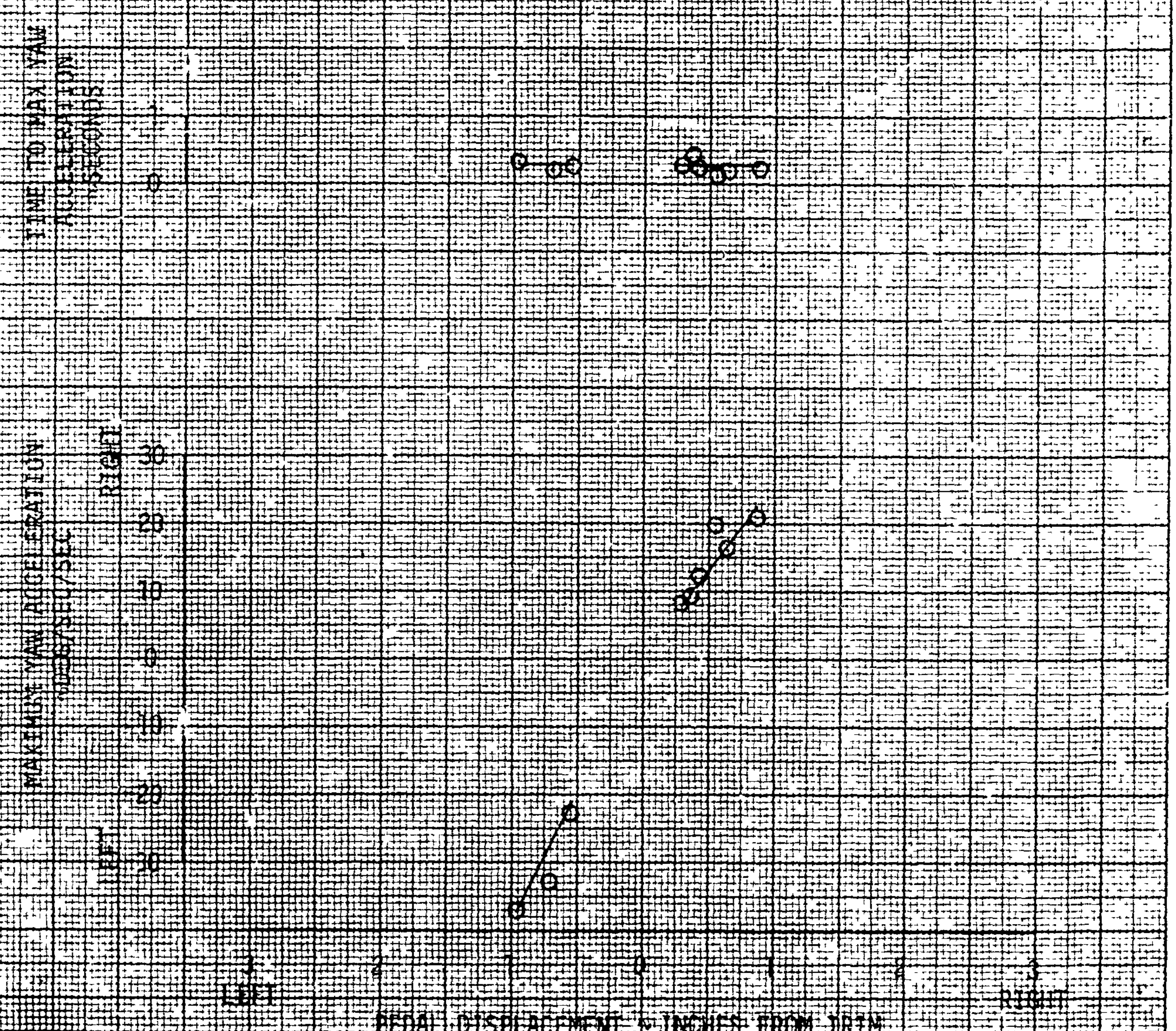


FIGURE 46
DIRECTIONAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,240	298.7	4320	248	8.3	62	CLEAN (GEAR DN)

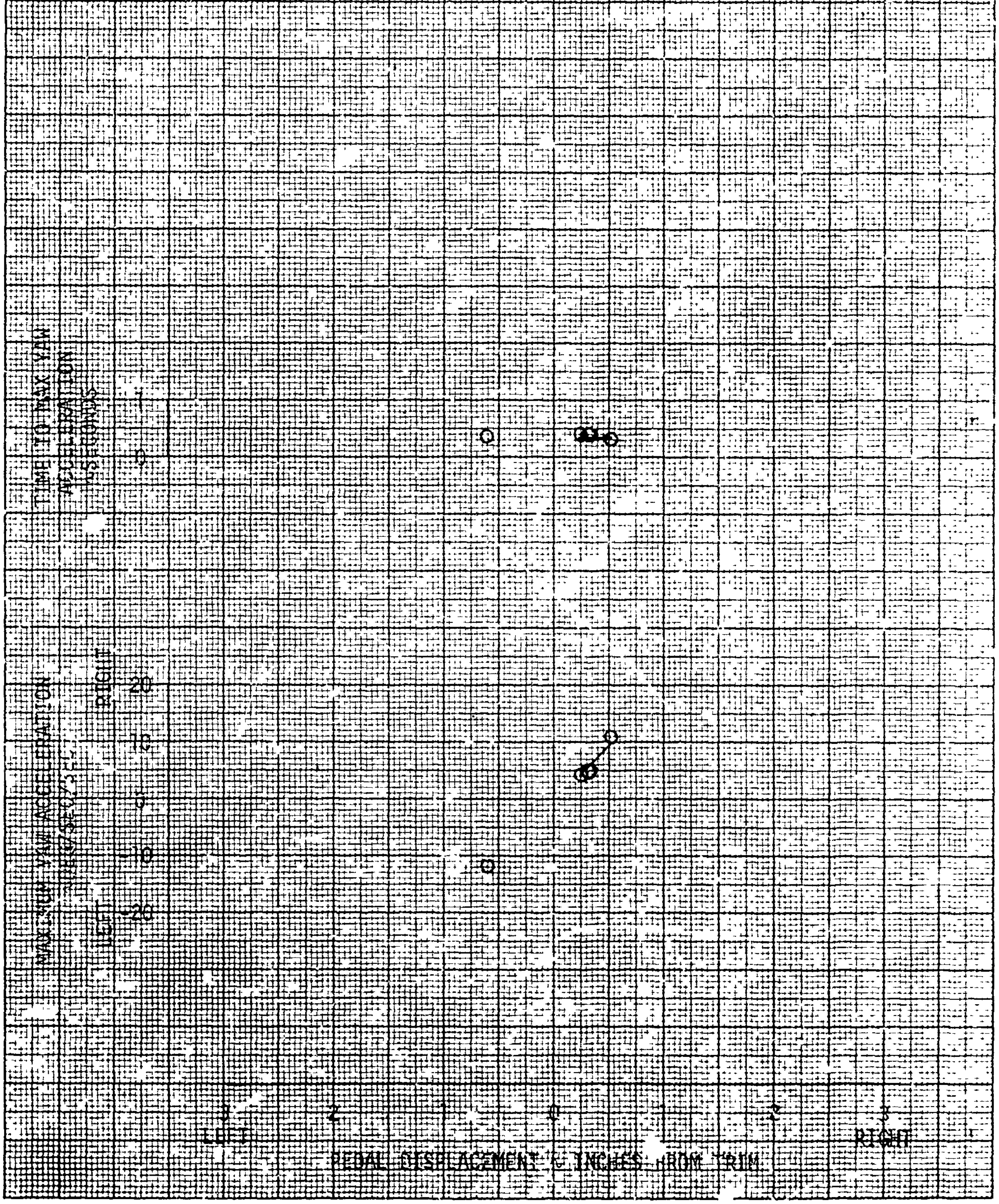


FIGURE 47
DIRECTIONAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CAL B AIRSPEED ~KT	CONFIGURATION
18,190	299.7	4020	247	7.0	101	CLEAN (GEAR UP)

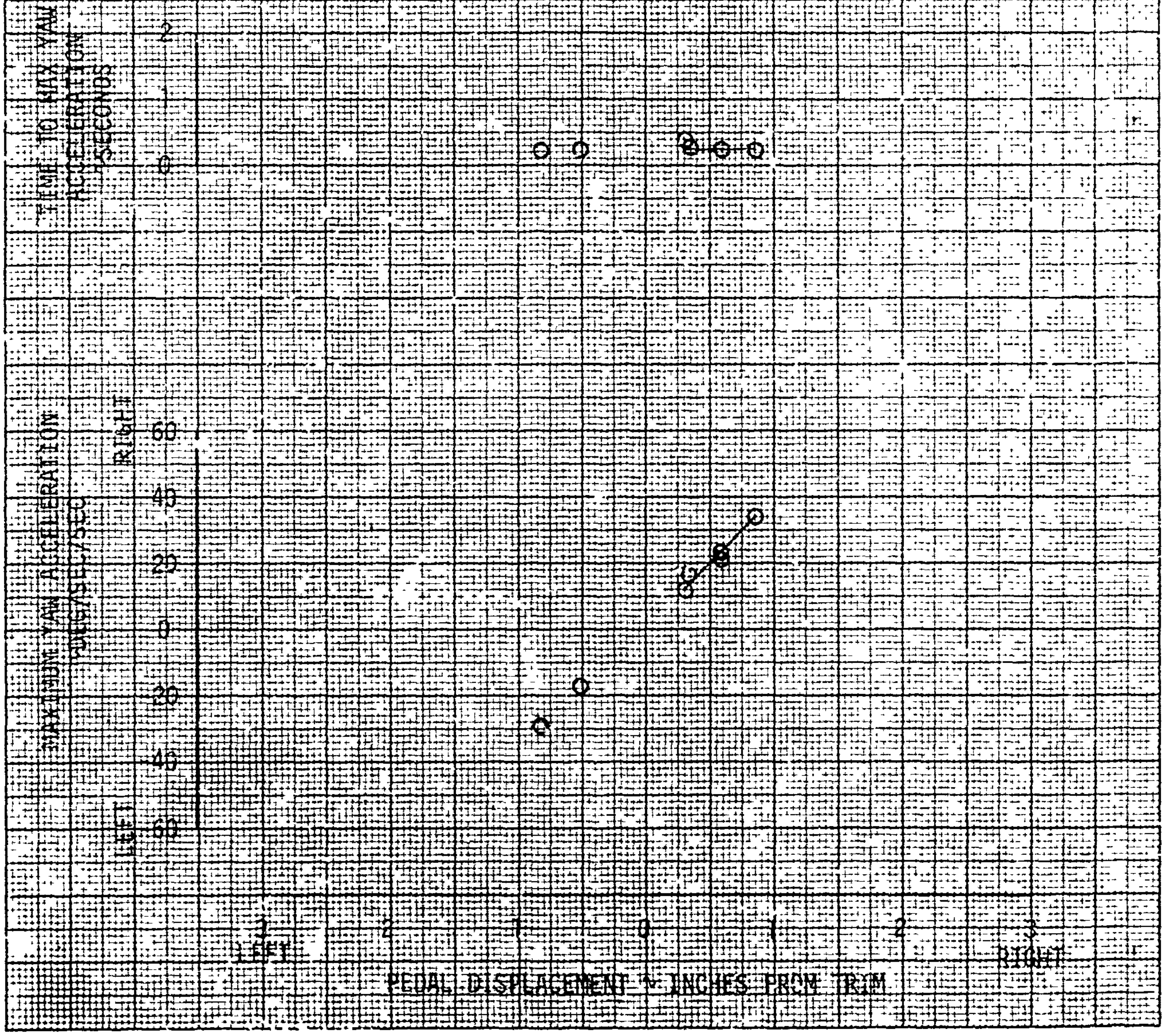


FIGURE 49
DIRECTIONAL CONTROL SENSITIVITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~%	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
18,010	259.3	5900	248	5.3	150	CLEAN (GEAR UP)

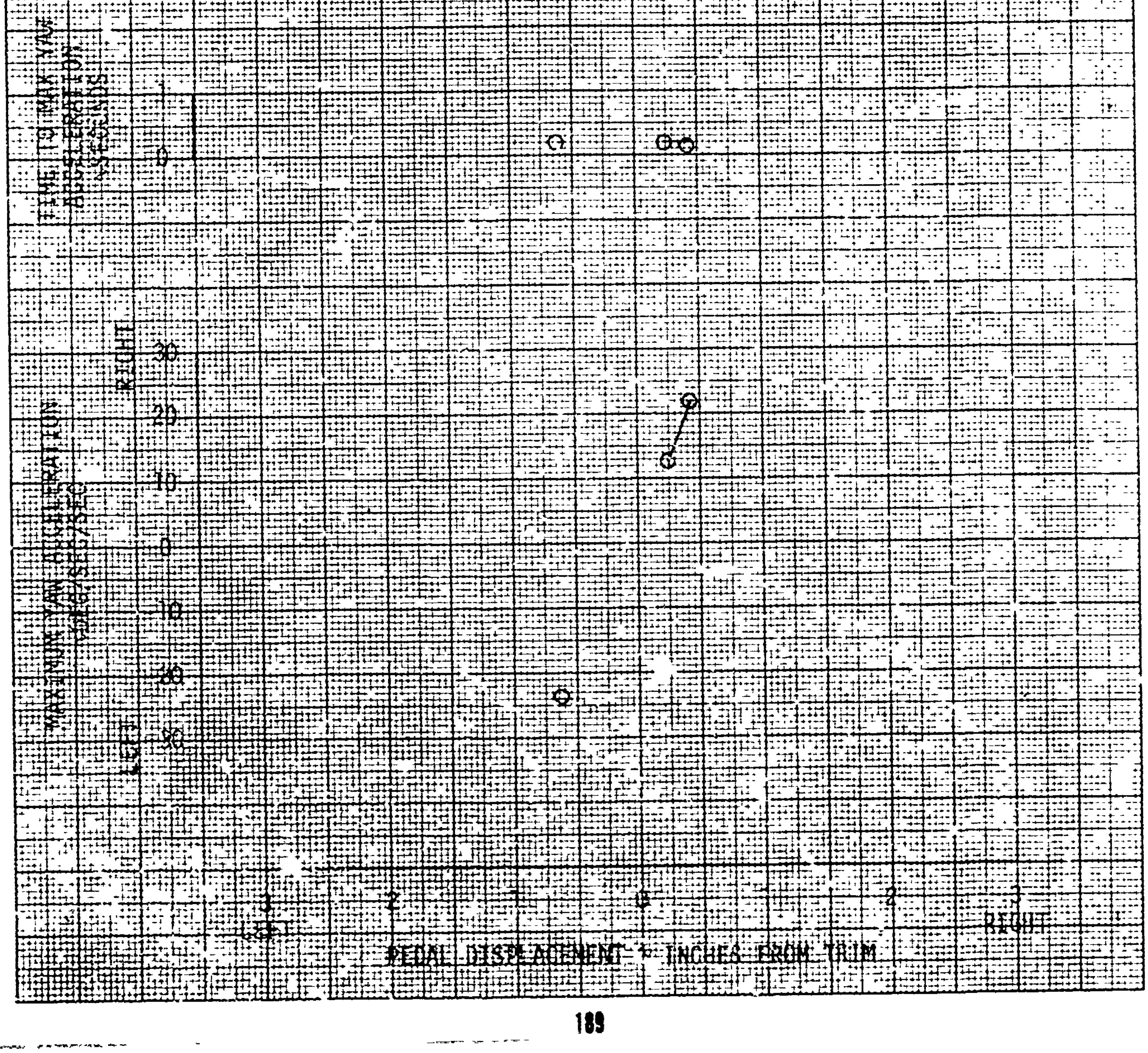


FIGURE 49

AFT LONGITUDINAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,546	298.9	-500	248	HOVER (IGE~15 FT)	11.70	-1.11	CLEAN (GEAR ON)

PITCH & C.G. ACCEL ———
ROLL — — —
YAW — - — -

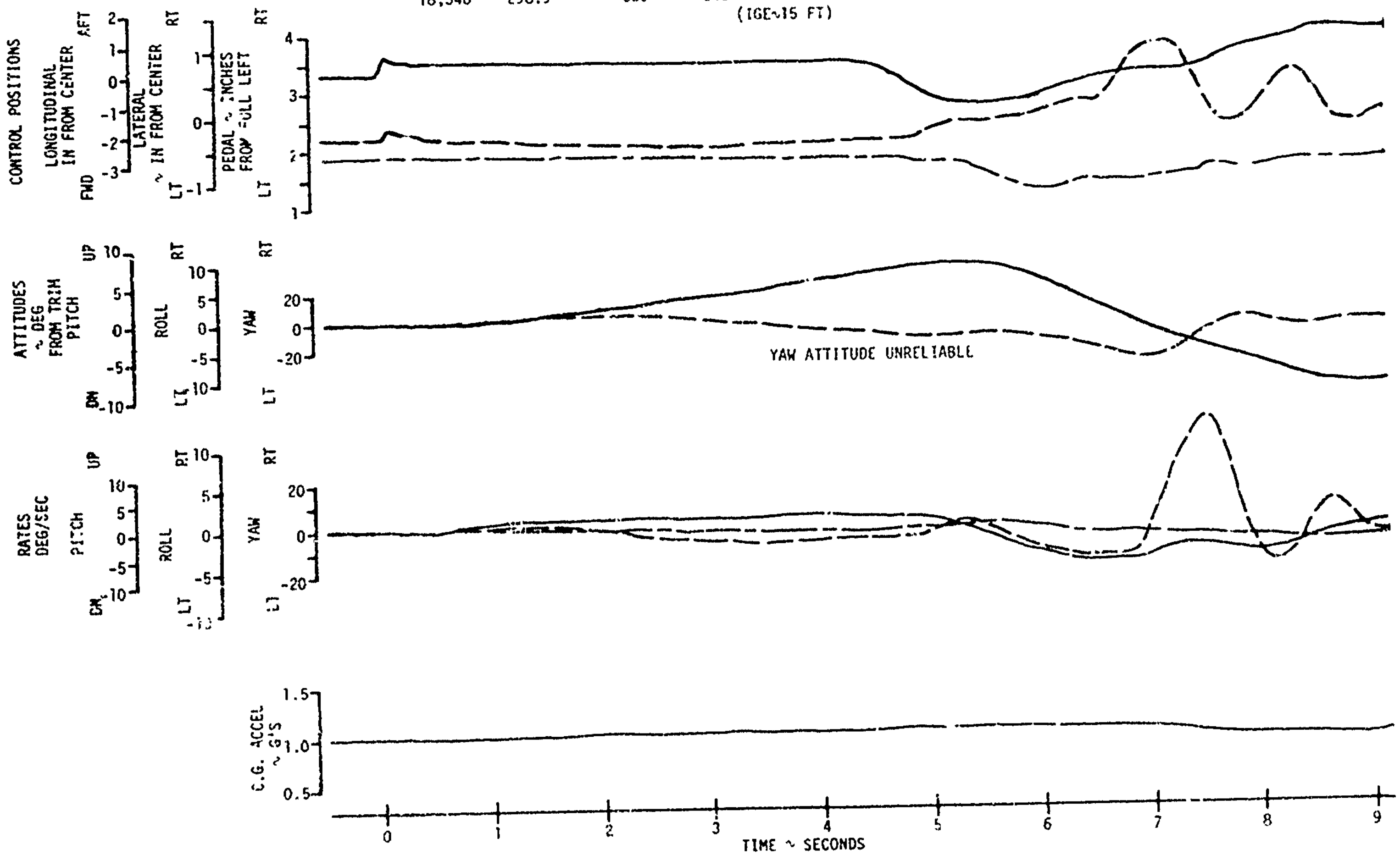


FIGURE 50

FORWARD LONGITUDINAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE DEG	CONFIGURATION
18,260	298.8	3340	247	64	8.2	13.(	CLEAN (GEAR ON)

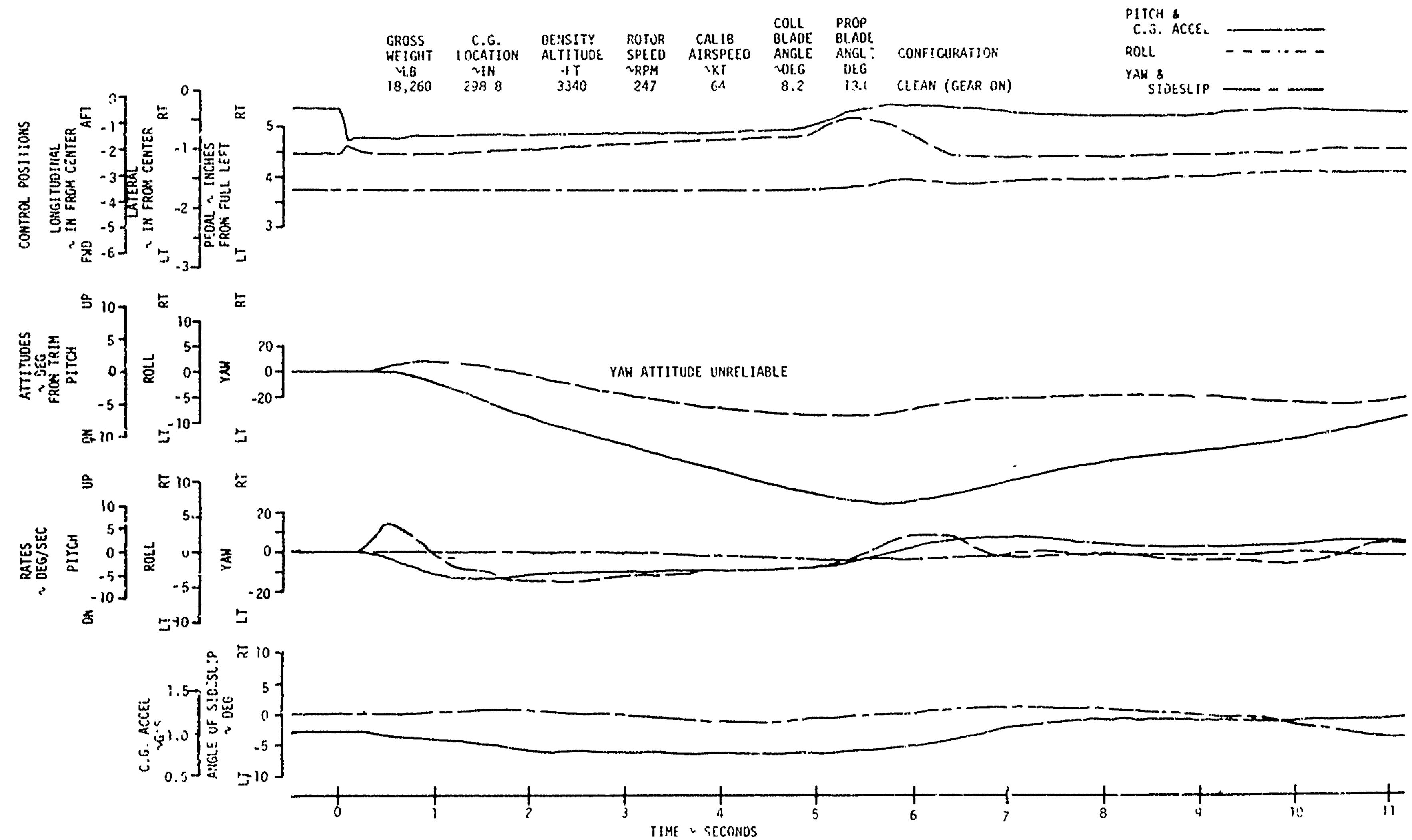


FIGURE 51

AFT LONGITUDINAL SILP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,450	299.8	4110	246	101	7.0	18.6	CLEAN (GEAR UP)

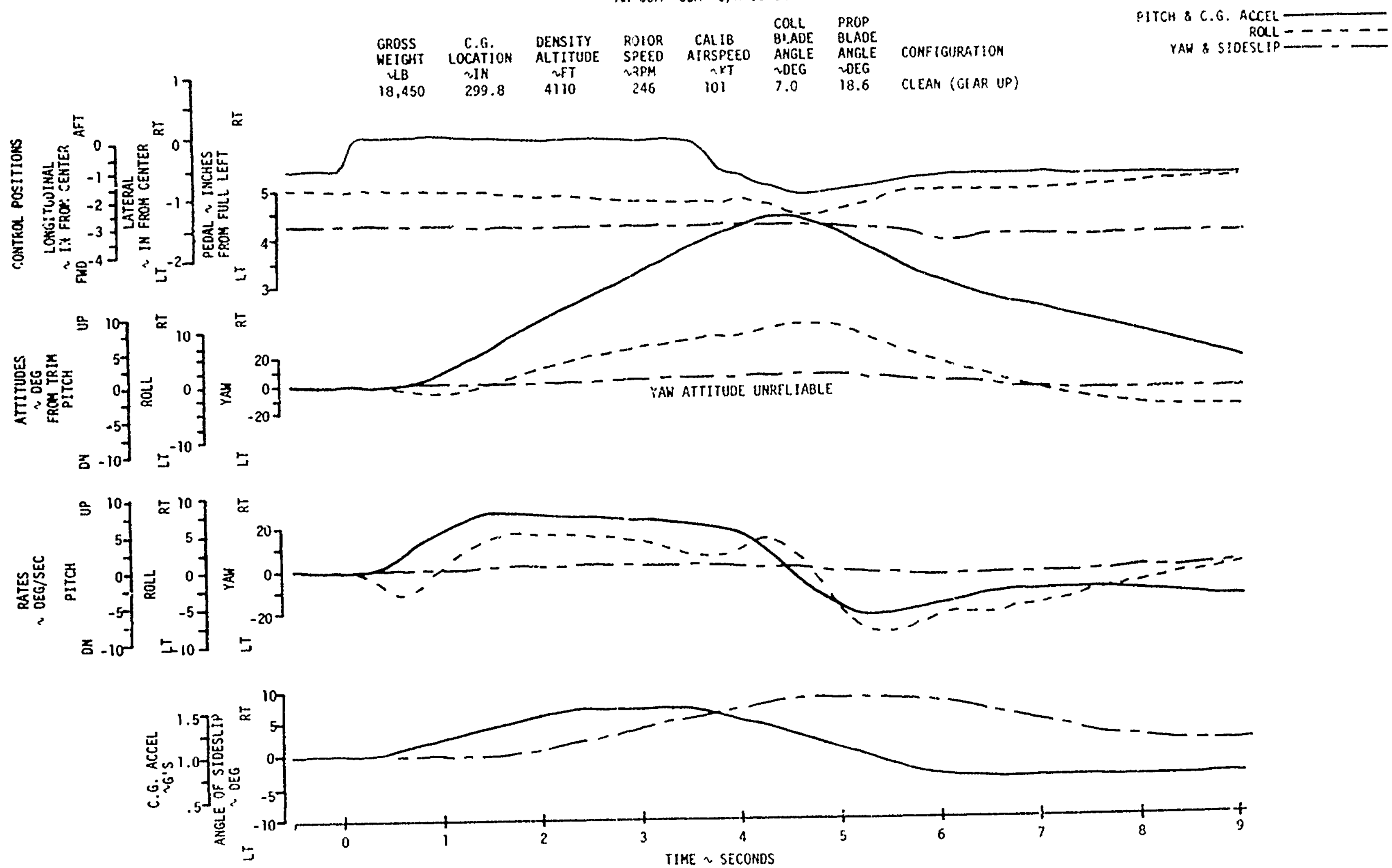


FIGURE 52

FORWARD LONGITUDINAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~ DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,550	299.8	3980	246	101	7.0	18.6	CLEAN (GEAR UP)

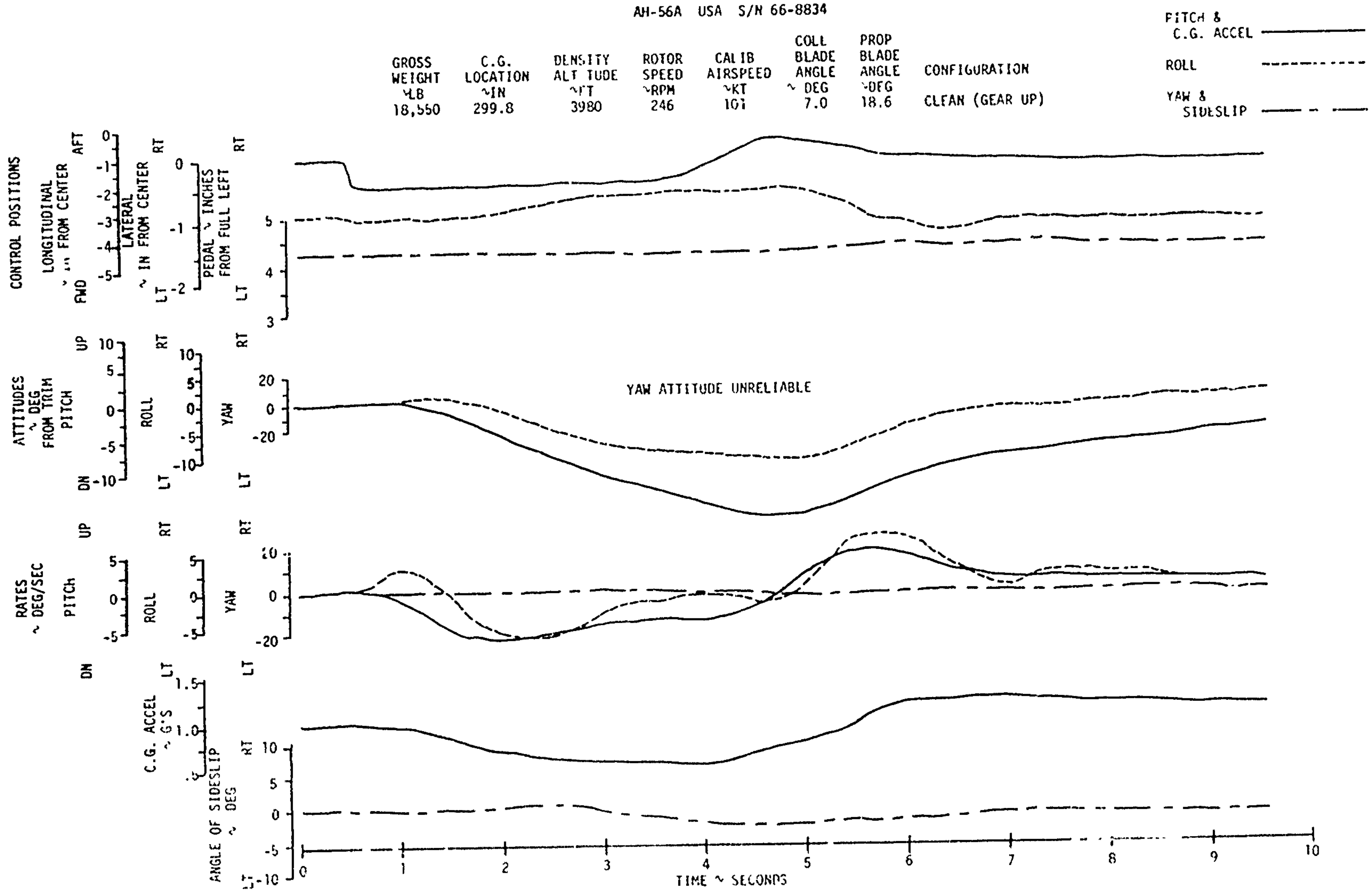


FIGURE 53
AFT LONGITUDINAL STEP
AH-56A USA S/N 66-8834

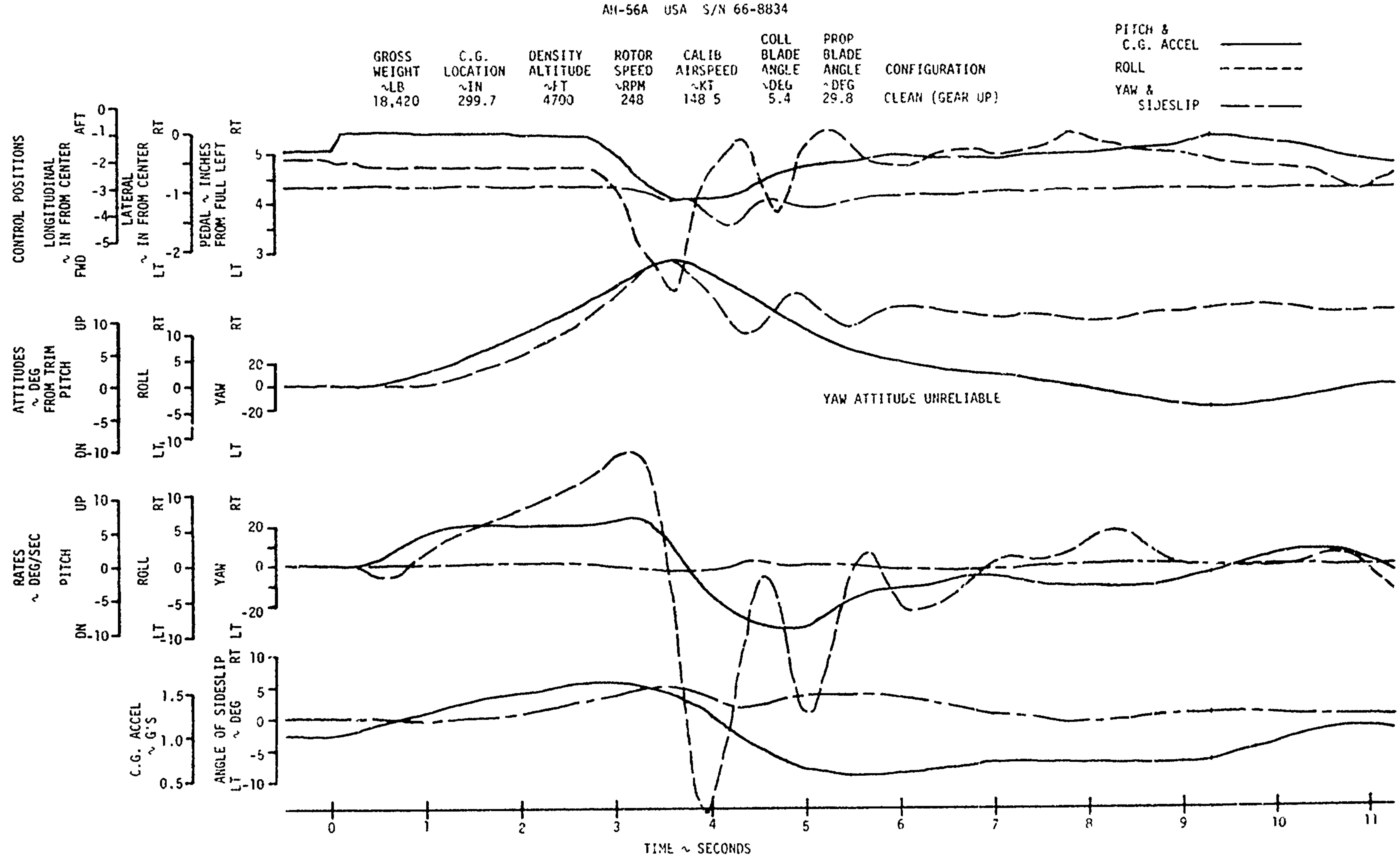


FIGURE 54
FORWARD LONGITUDINAL STEP
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,020	299.6	4360	248	149	5.6	29.6	CLEAN (GEAR UP)

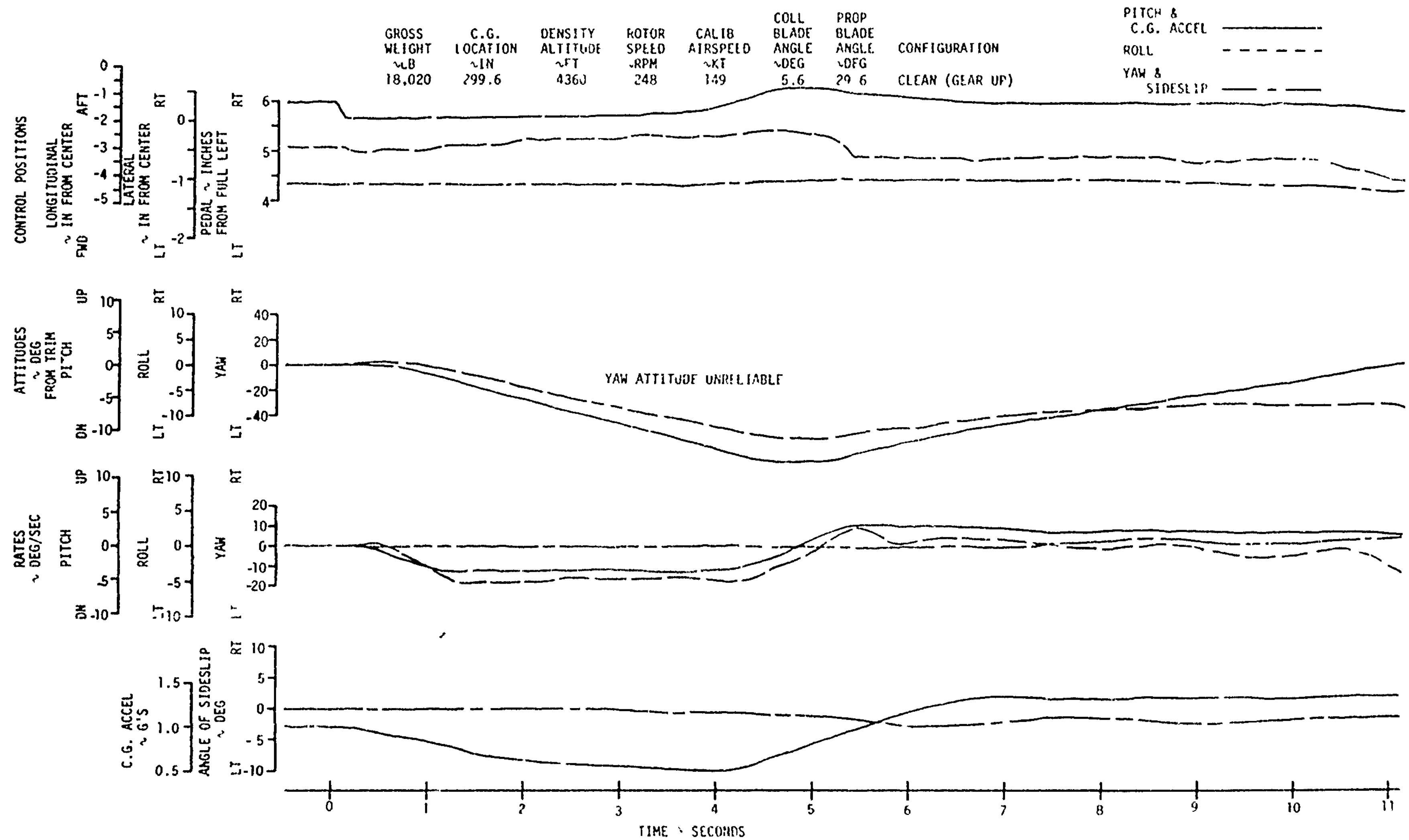


FIGURE 55
RIGHT LATERAL STEP
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE DEG	PROP BLADE ANGLE DEG	CONFIGURATION
18,270	298.8	-500	246	HOVER	11.2	-1.6	CLEAN (GEAR DN)

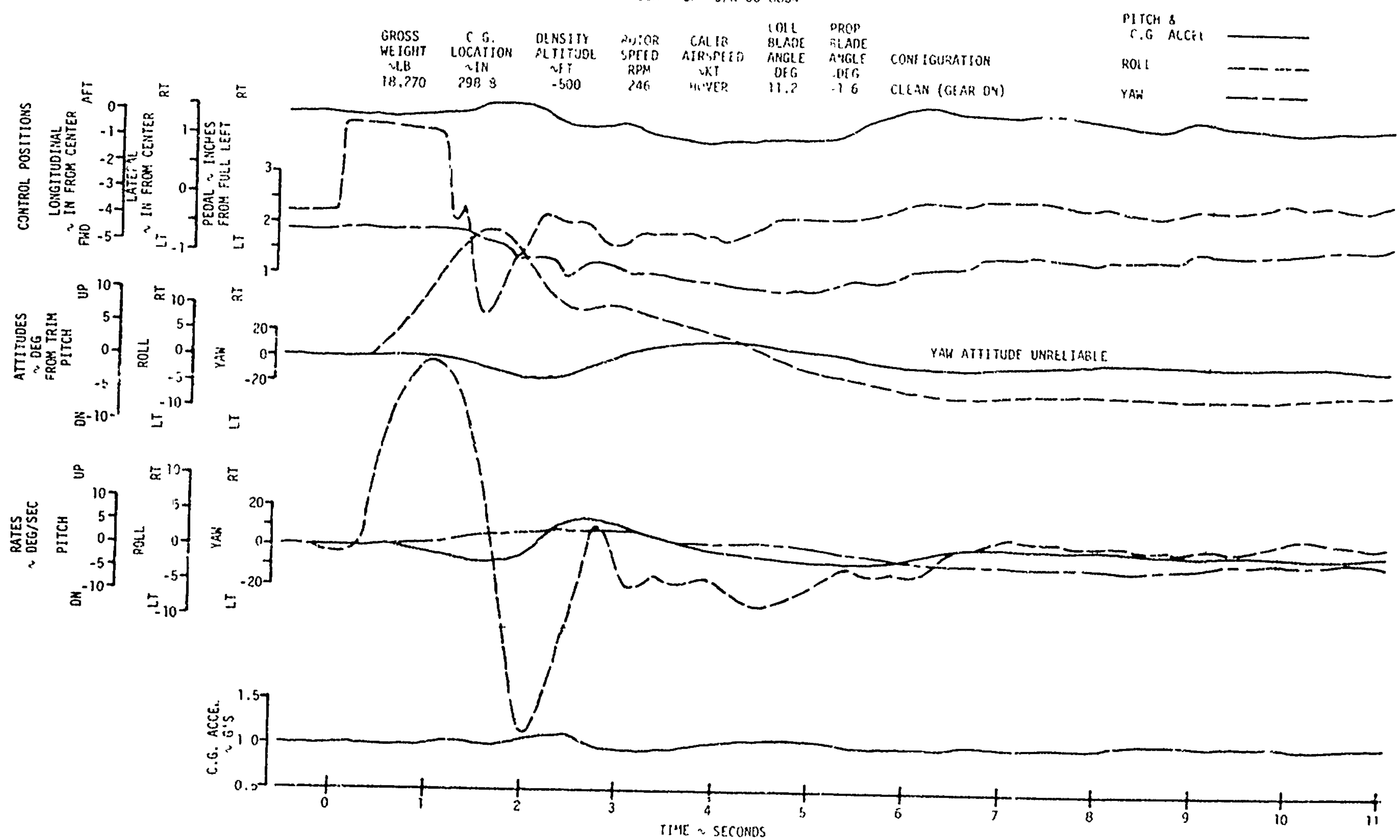


FIGURE 56
LEFT LATERAL STE.
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,270	298.8	-500	246	HOVER	11.2	-1.6	CLEAN (GEAR ON)

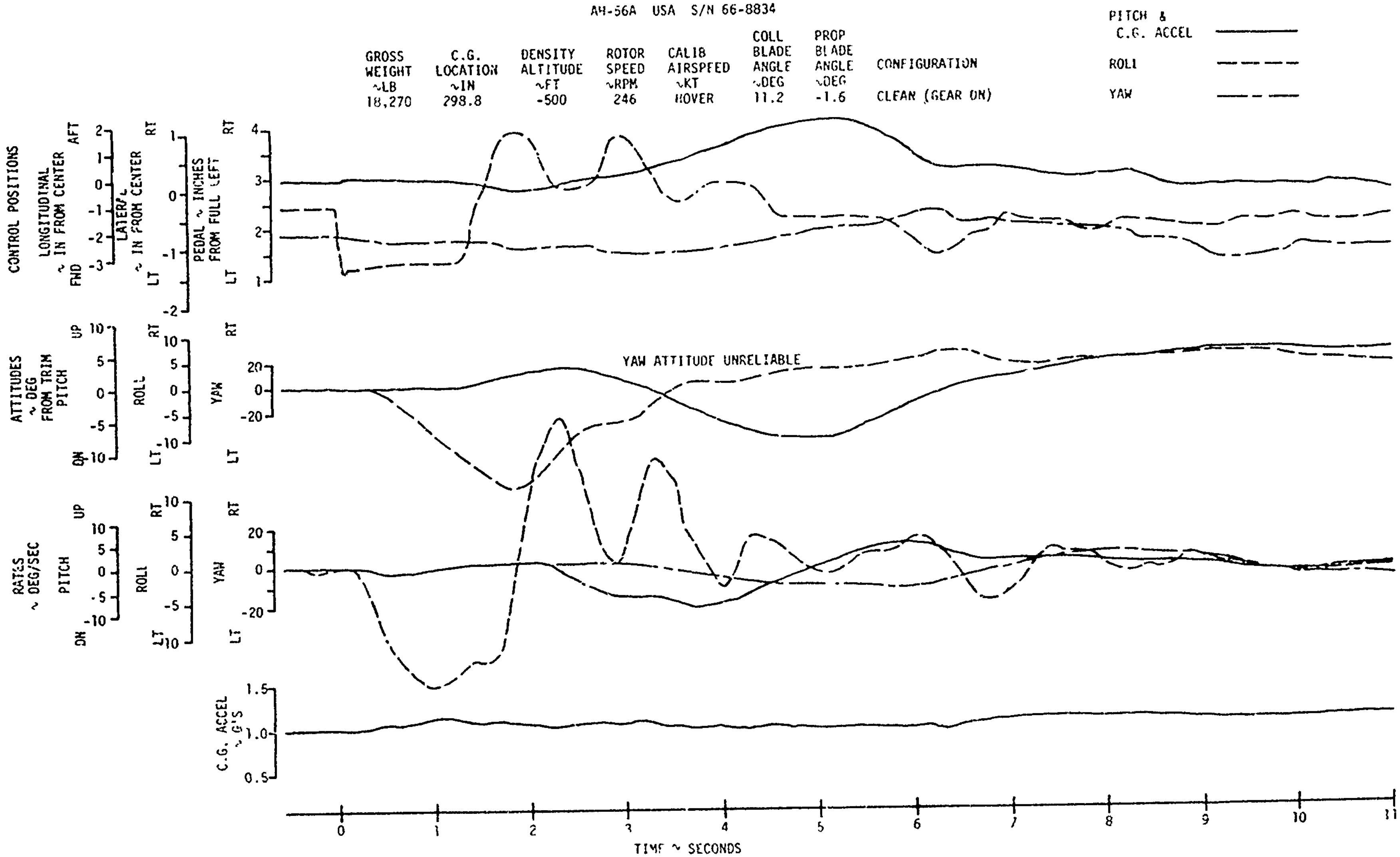


FIGURE 57

RIGHT LATERAL

AH-56A USA S, 5834

GROSS WEIGH ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,190	298.7	3470	249	63	8.3	13.0	CLEAN (GEAR ON)

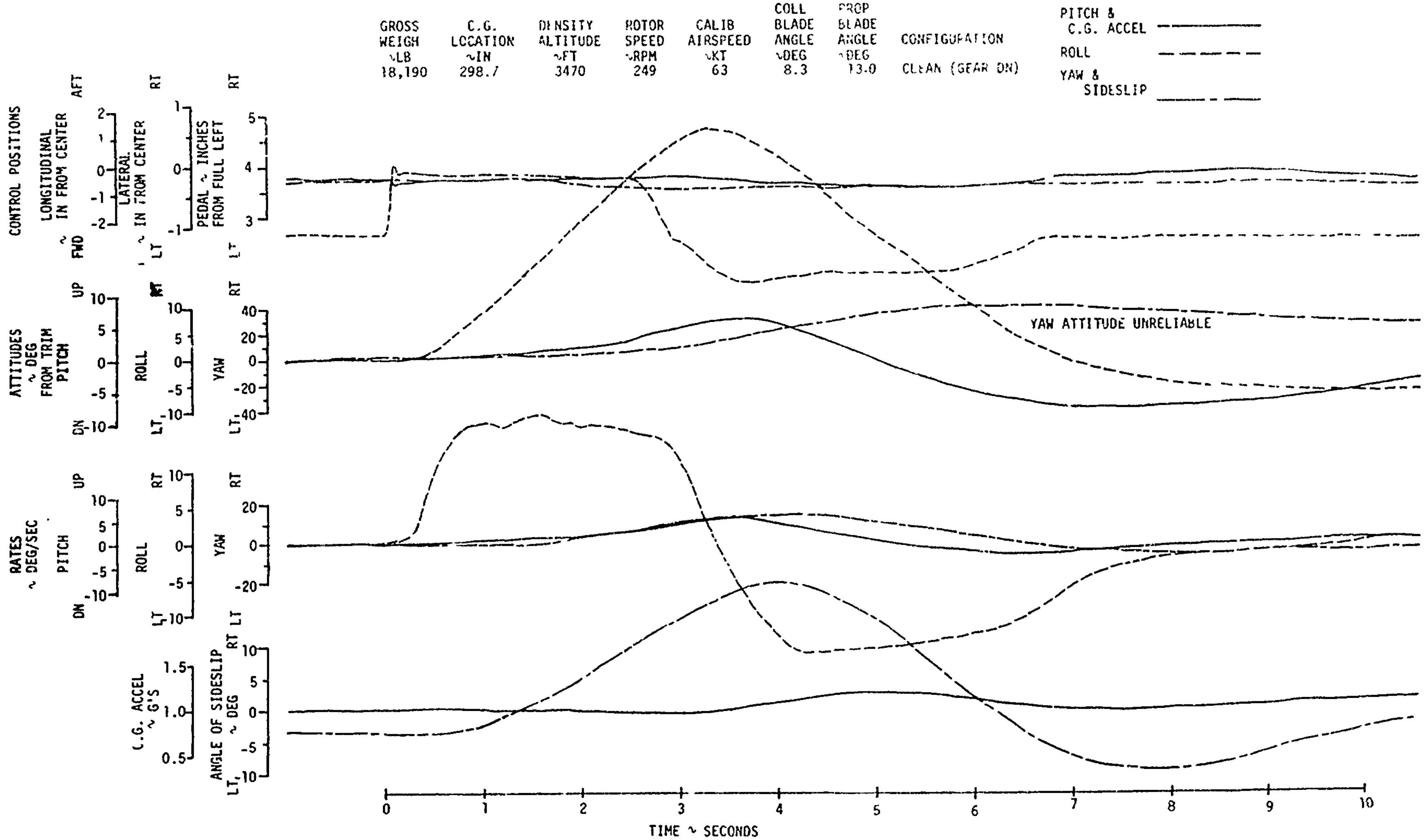


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FIGURE 58

LEFT LATERAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,110	298.7	3160	247	63.5	8.1	13.2	CLEAN (GEAR ON)

PITCH & C.G. ACCEL ————

ROLL — — — —

YAW & SIDESLIP — · — · —

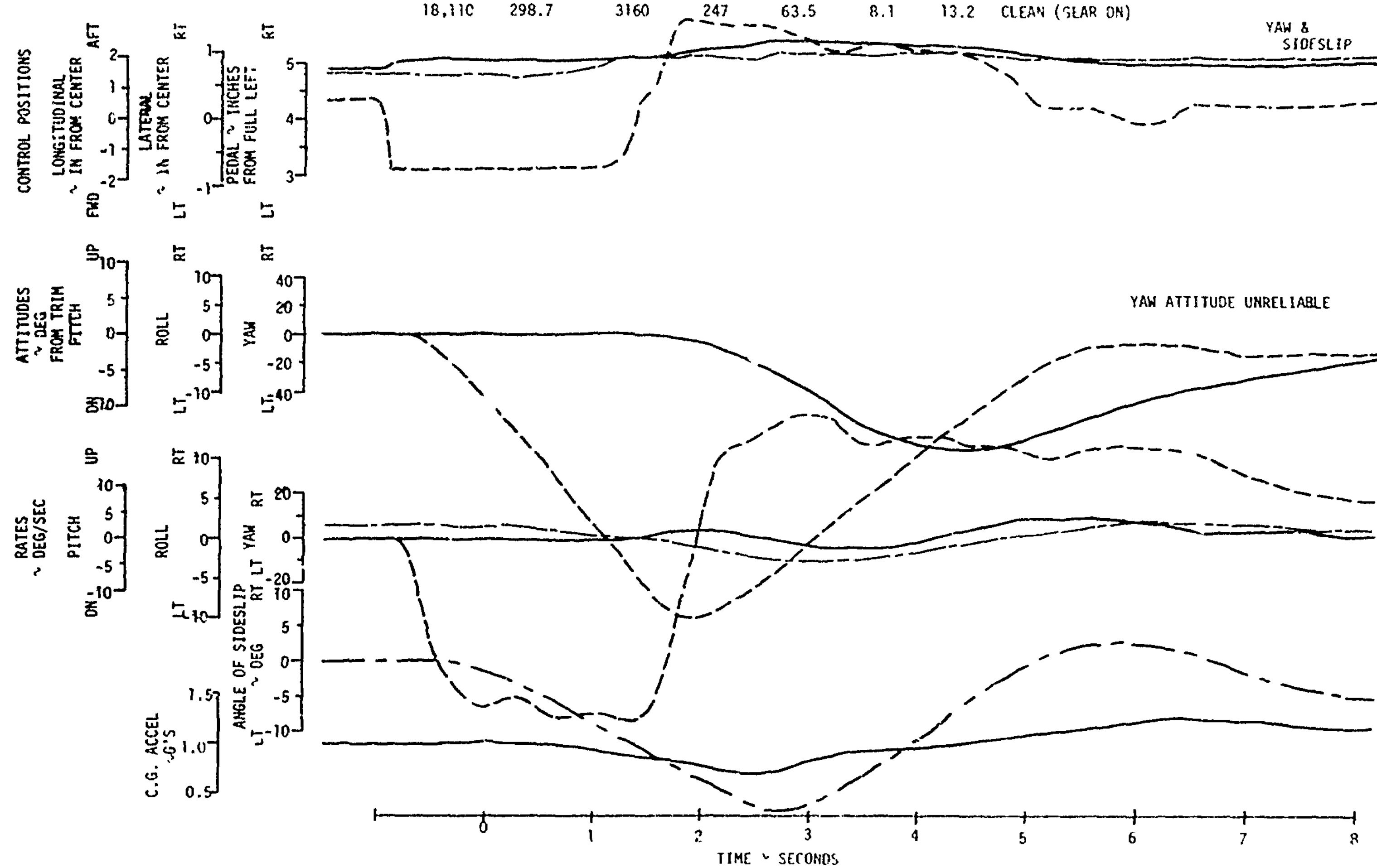


FIGURE 59

RIGHT LATERAL STEP
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,370	299.7	3060	247	100	.0	18.6	CLEAN (GEAR UP)

PITCH & C.G. ACCEL ————
ROLL - - - - - -
YAW & SIDESLIP —— - ——

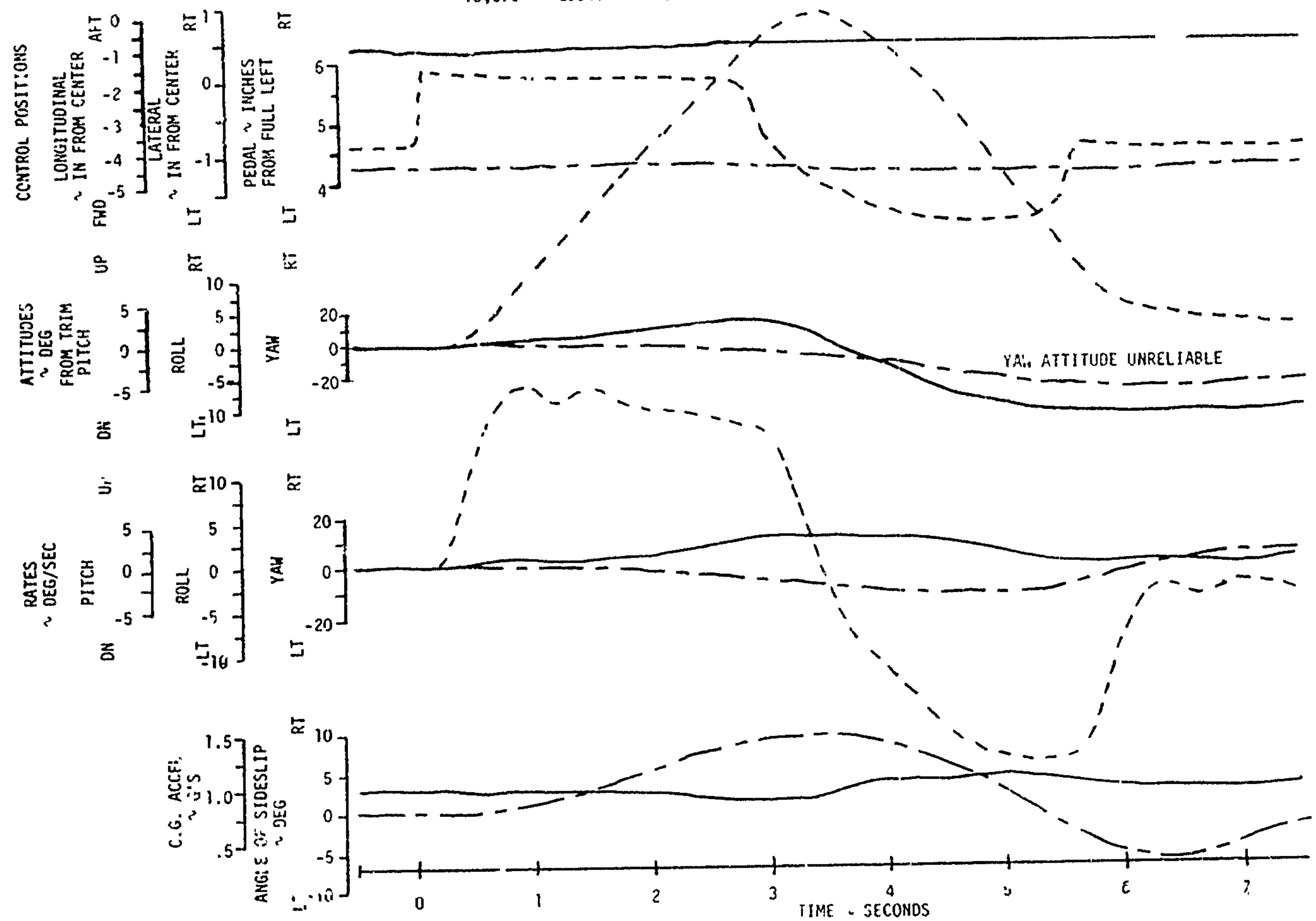


FIGURE 60

RIGHT LATERAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
17,640	299.6	4800	248	149	5.6	29.6	CLEAN (GEAR UP)

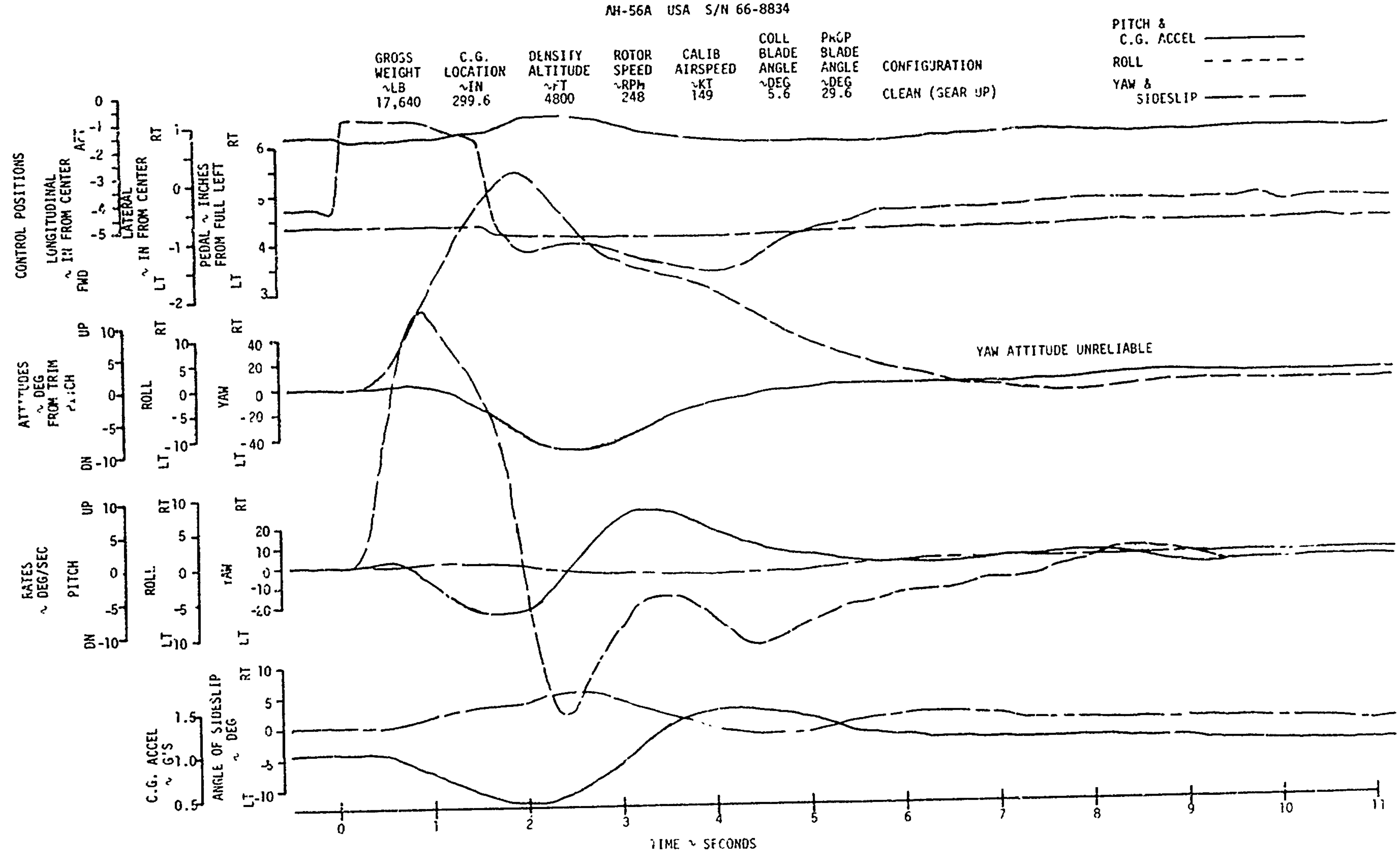


FIGURE 61

LEFT LATERAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
17,590	299.8	4720	248	150	5.6	29 6	CLEAN (GEAR UP)

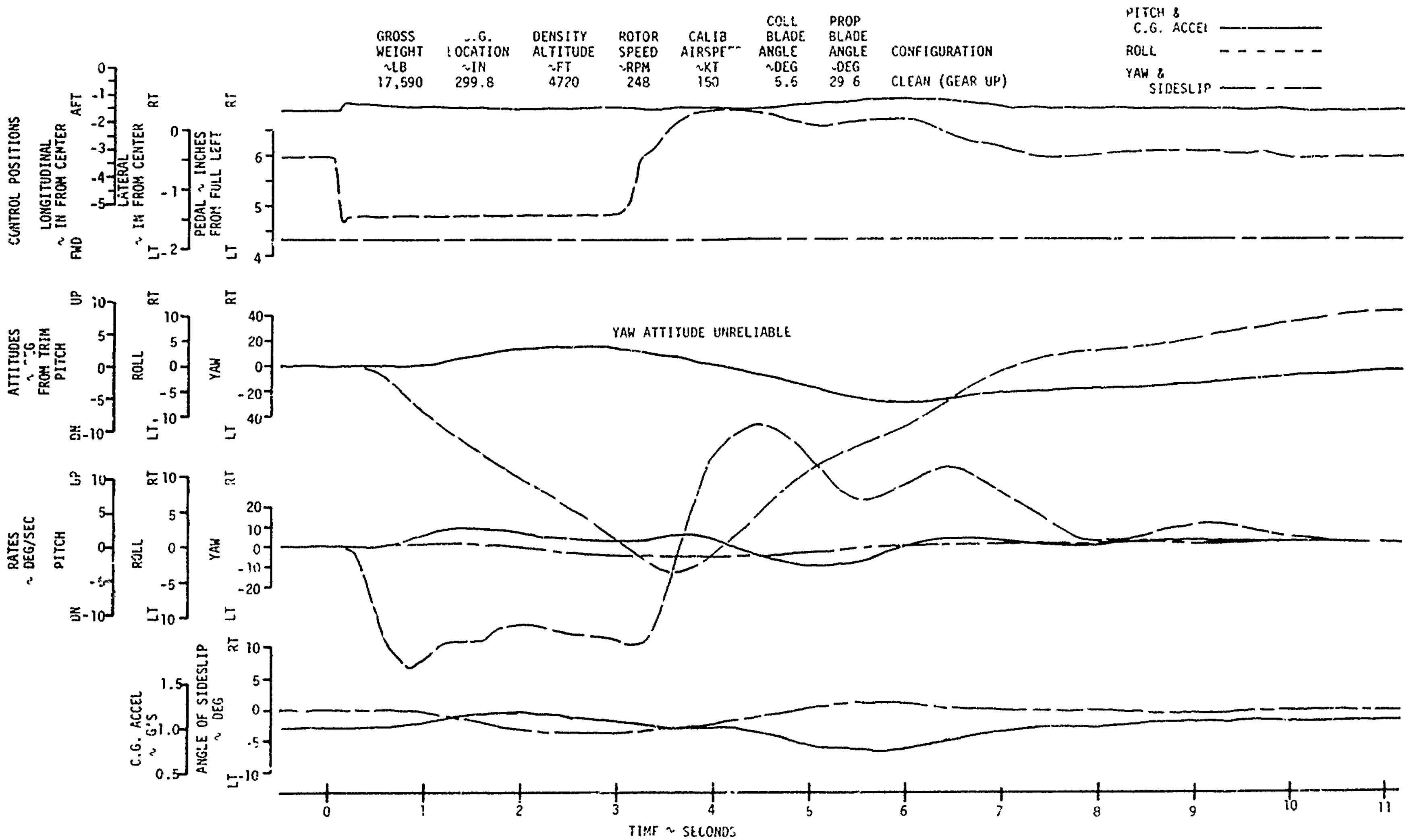


FIGURE 62

RIGHT LATERAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,340	299.7	5900	248	167	5.1	34.4	CLEAN (GEAR UP)

PITCH & C.G. ACCEL ————
ROLL – – – – –
YAW & SIDESLIP —— – · ——

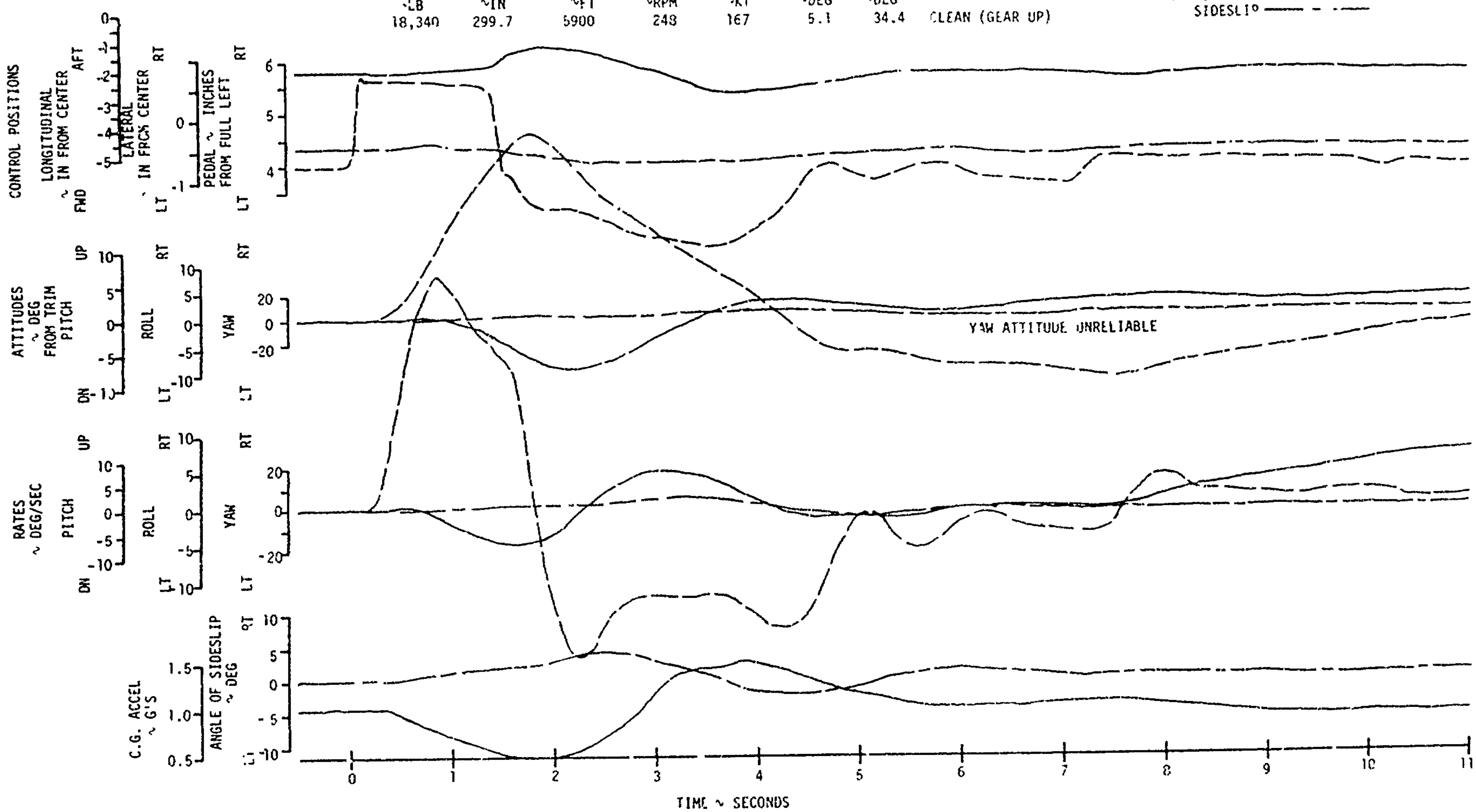


FIGURE 63

LEFT LATERAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,250	299.7	5690	248	166	5.1	34.3	CLEAN (GEAR UP)

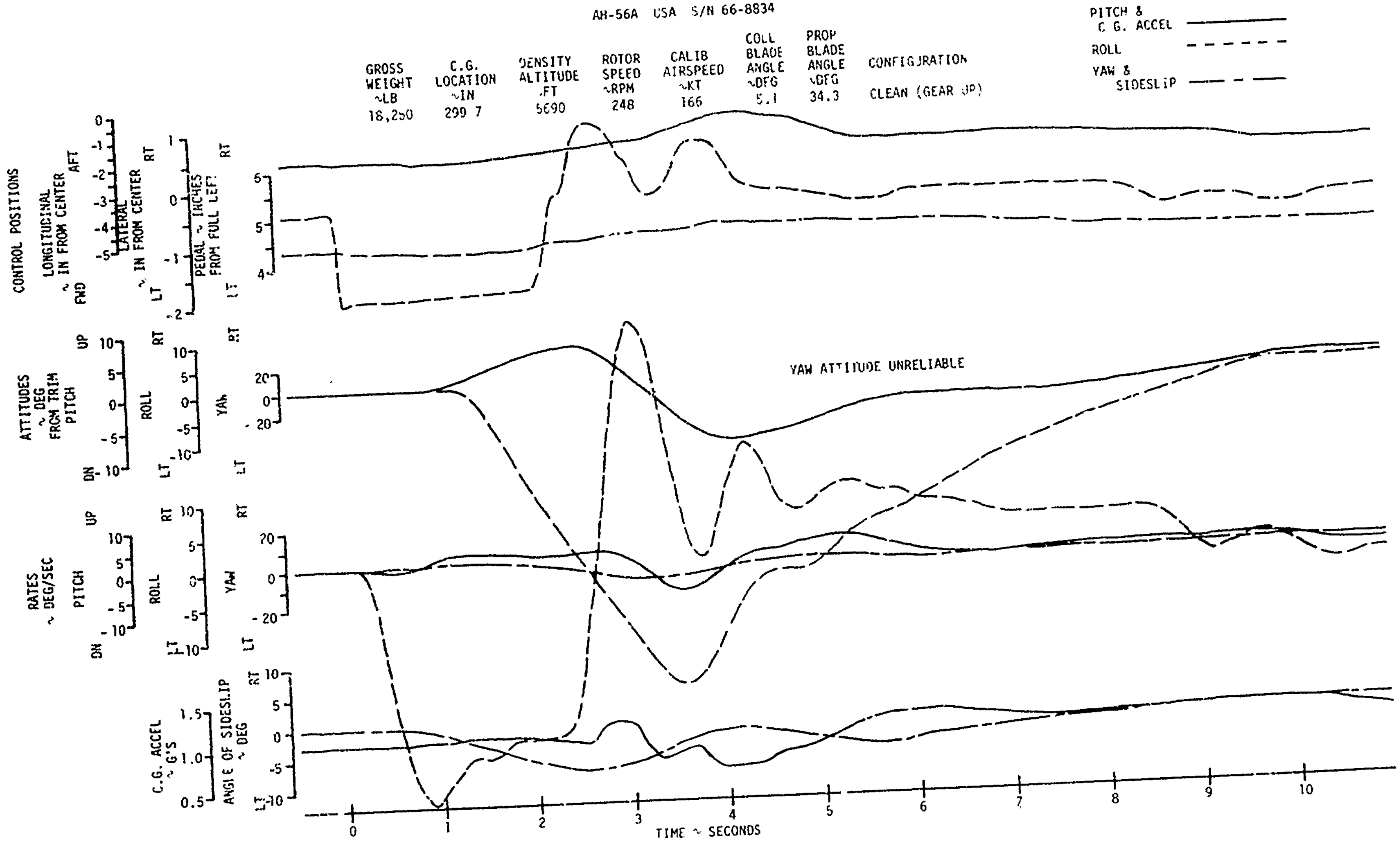


FIGURE 64

RIGHT LATERAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,300	298.7	3760	249	179.5	5.1	31.0	CLEAN (GEAR UP)

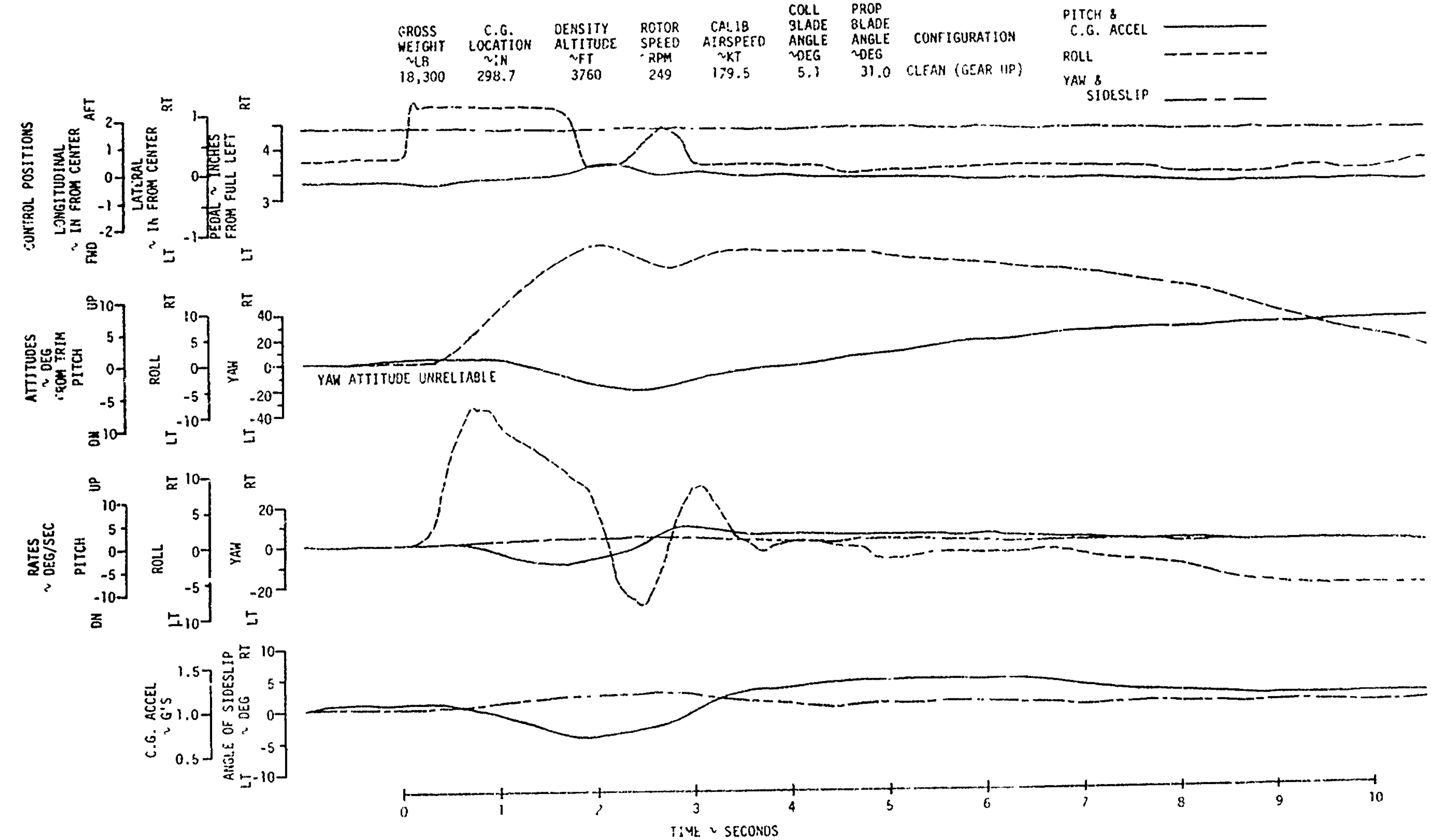


FIGURE 65

LEFT LATERAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CAL'D AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,300	298.7	3760	248	179	5.1	31 0	CLEAN (GEAR UP)

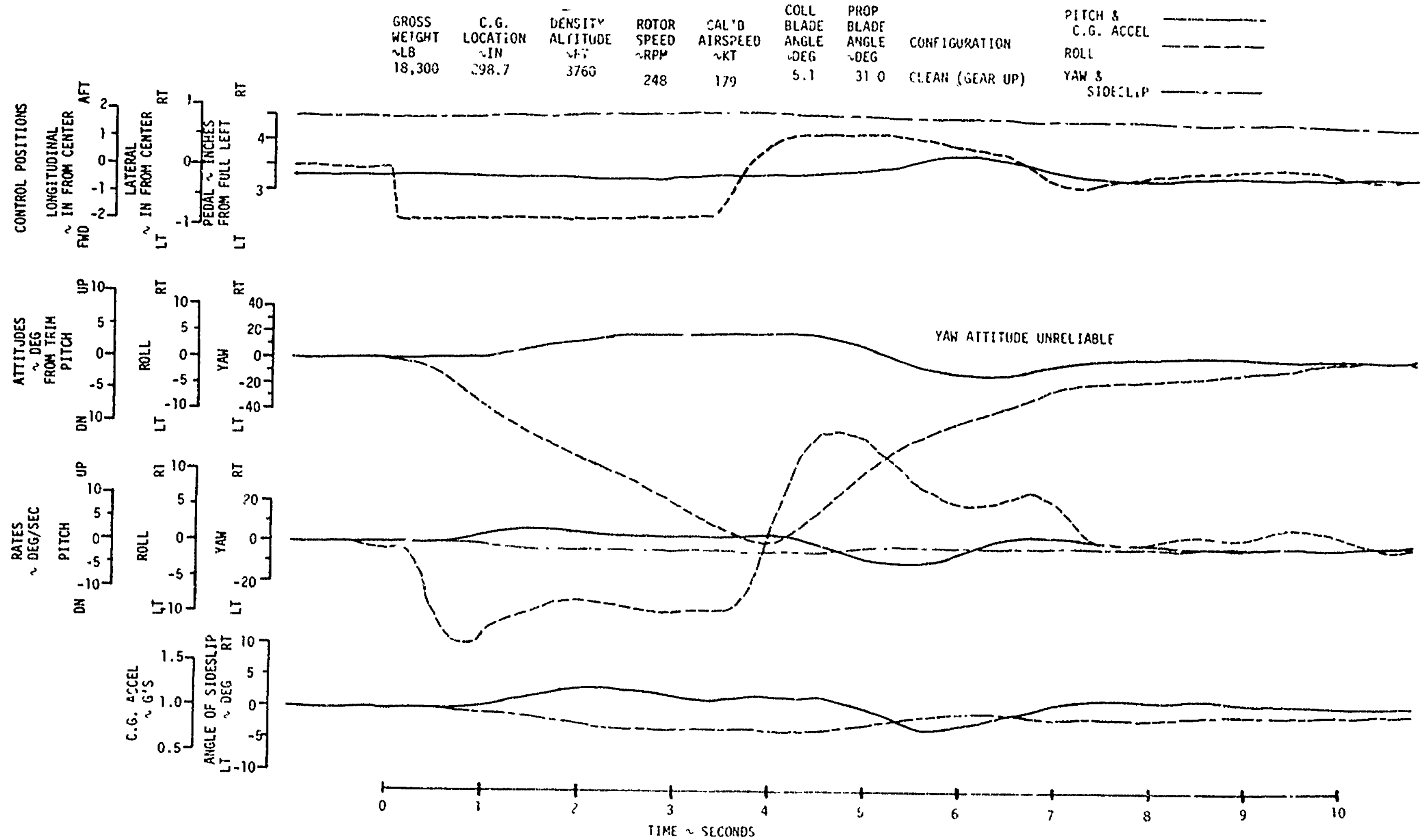


FIGURE 66

RIGHT PEDAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
17,980	298.7	-500	248	HOVER	11.3	-1.4	CLEAN (GEAR ON)

PITCH & C.G. ACCEL ———
ROLL — — —
YAW —— ——

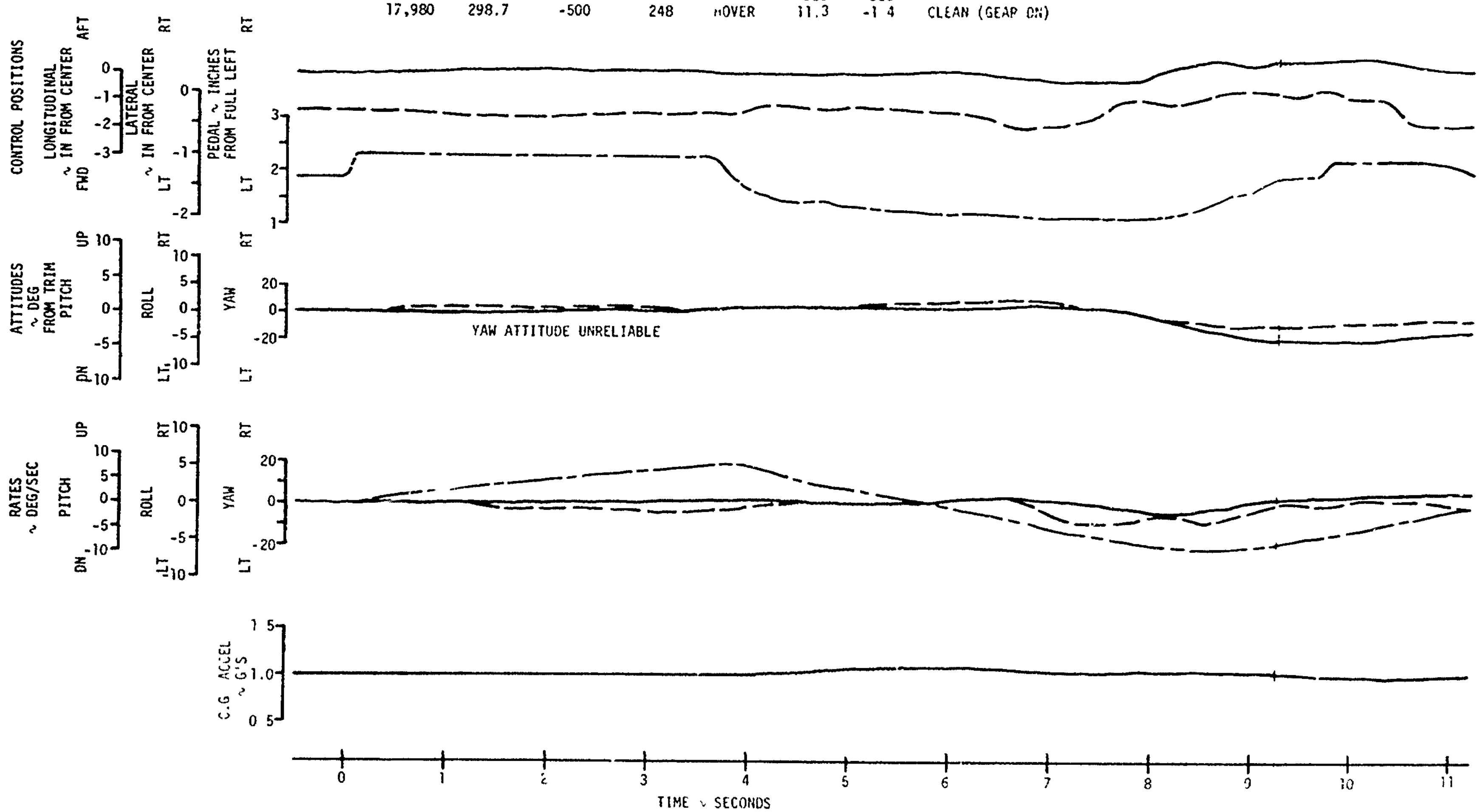


FIGURE 67

RIGHT PEDAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,020	298.7	4460	248	63	8.3	13.2	CLEAN (GEAR ON)

PITCH & C.G. ACCEL ————
ROLL - - - - - -
YAW & SIDESLIP —— - ——

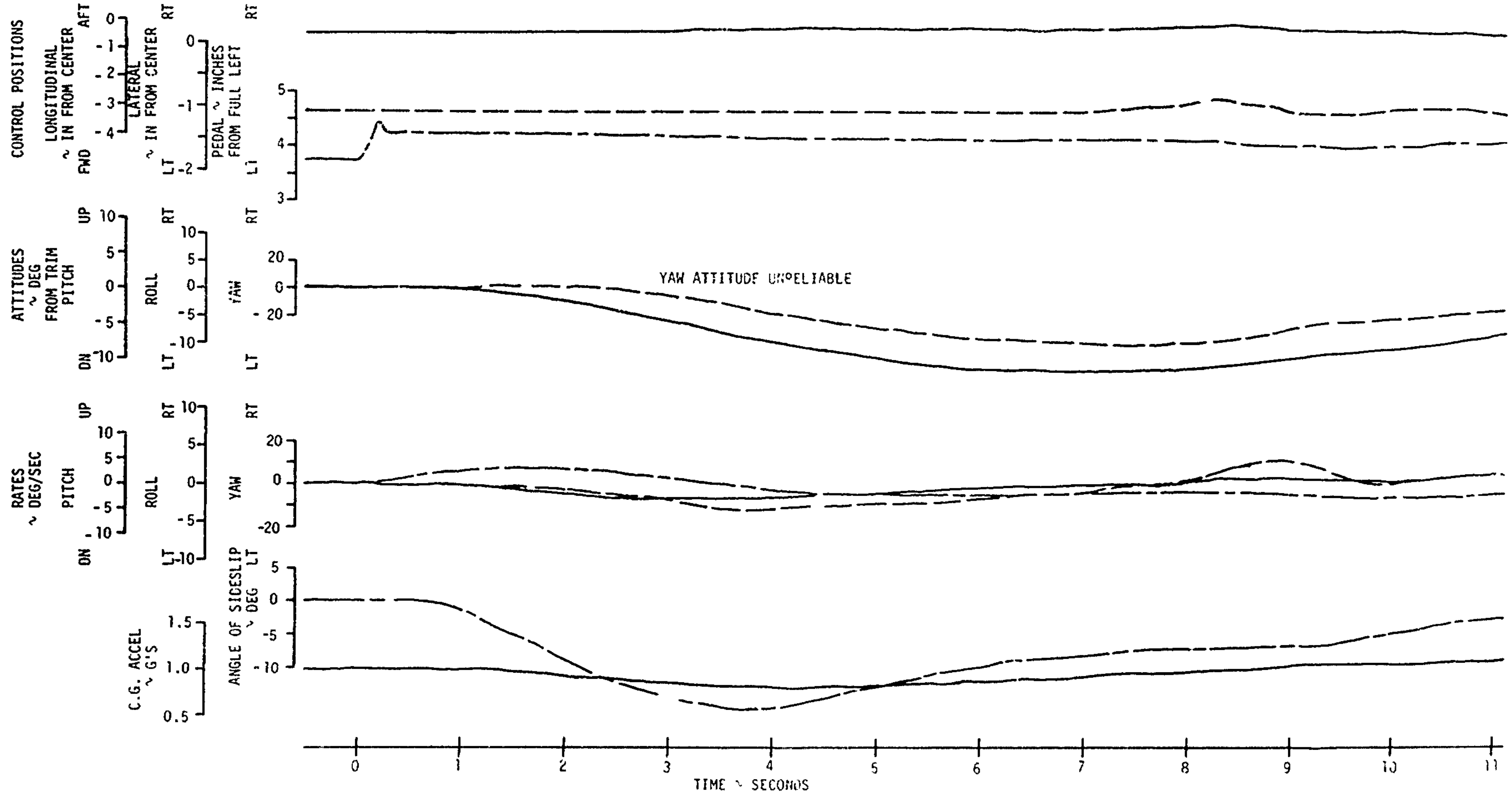


FIGURE 68

RIGHT PEDAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,250	299.7	3950	246	101	7.0	18.6	CLFAN (GEAR UP)

PITCH & C.G. ACCEL ————
ROLL - - - - - -
YAW & SIDESLIP — — —

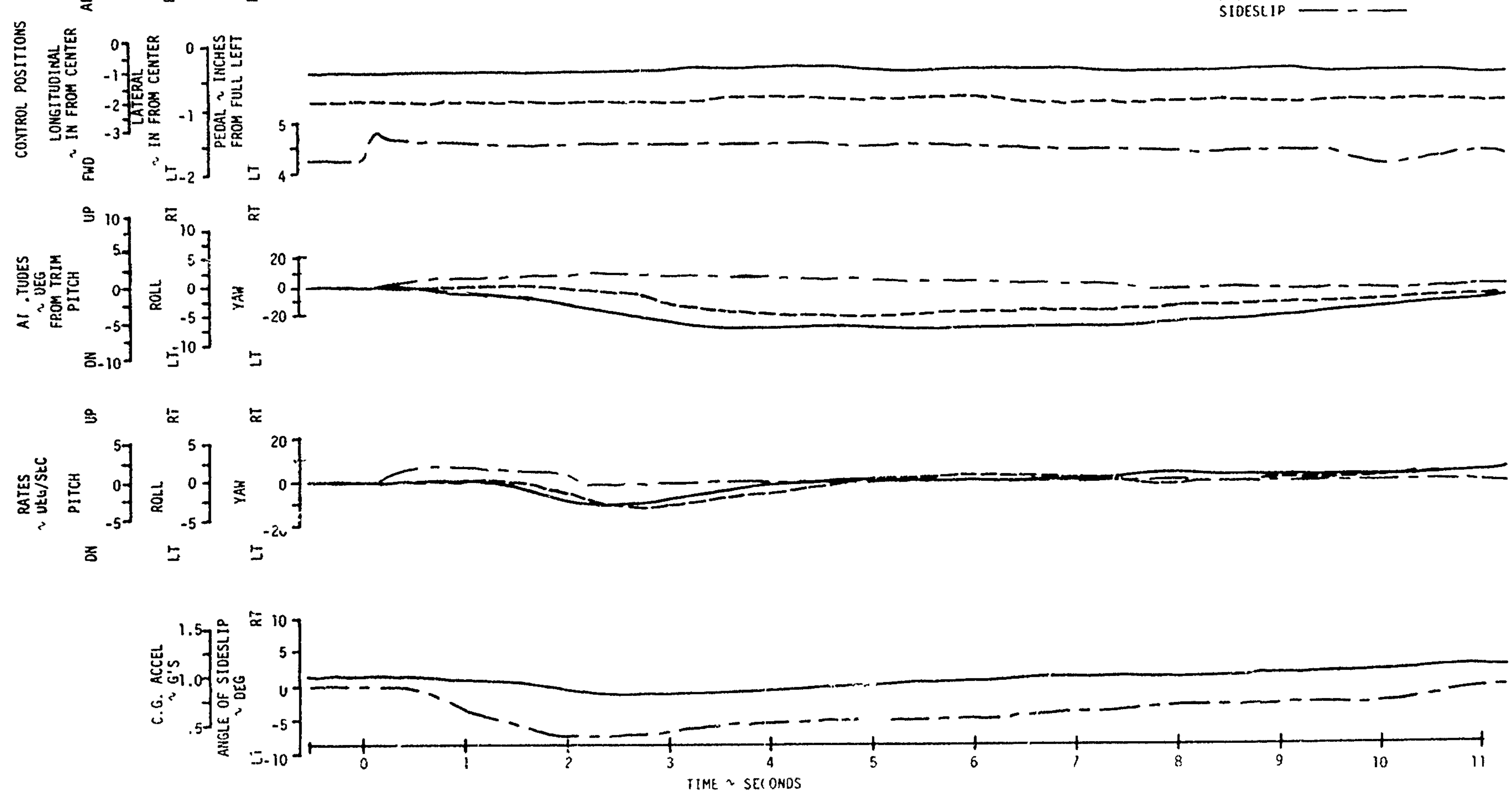


FIGURE 69

LEFT PEDAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,140	299.6	3950	246	101	7.0	18.6	CLEAN (GEAR UP)

PITCH & C.G. ACCEL ———
ROLL – – – –
YAW & SIDESLIP — – —

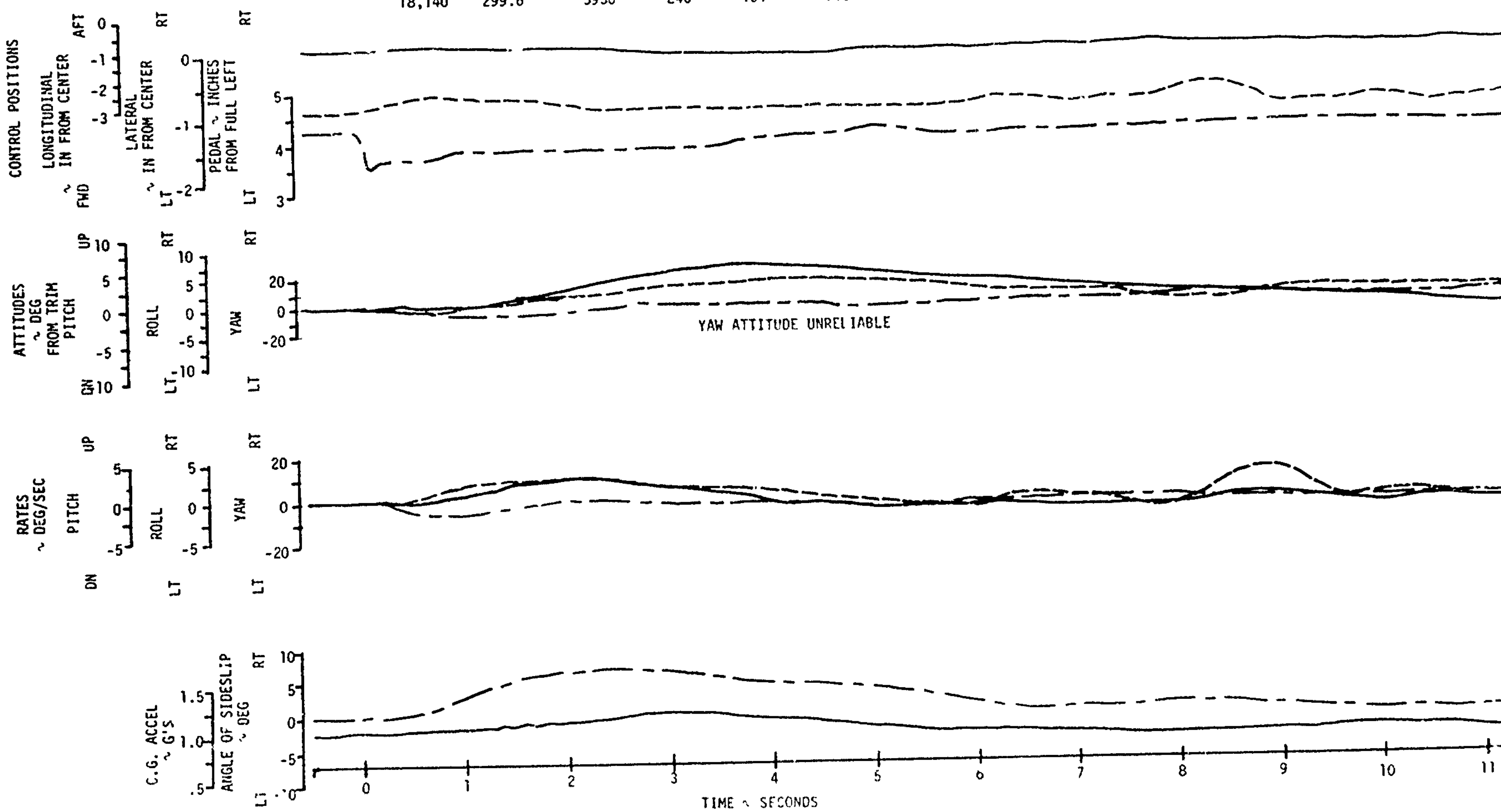


FIGURE 70

RIGHT PEDAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,020	299.6	5750	248	149.5	5.3	31.0	CLEAN (GEAR UP)

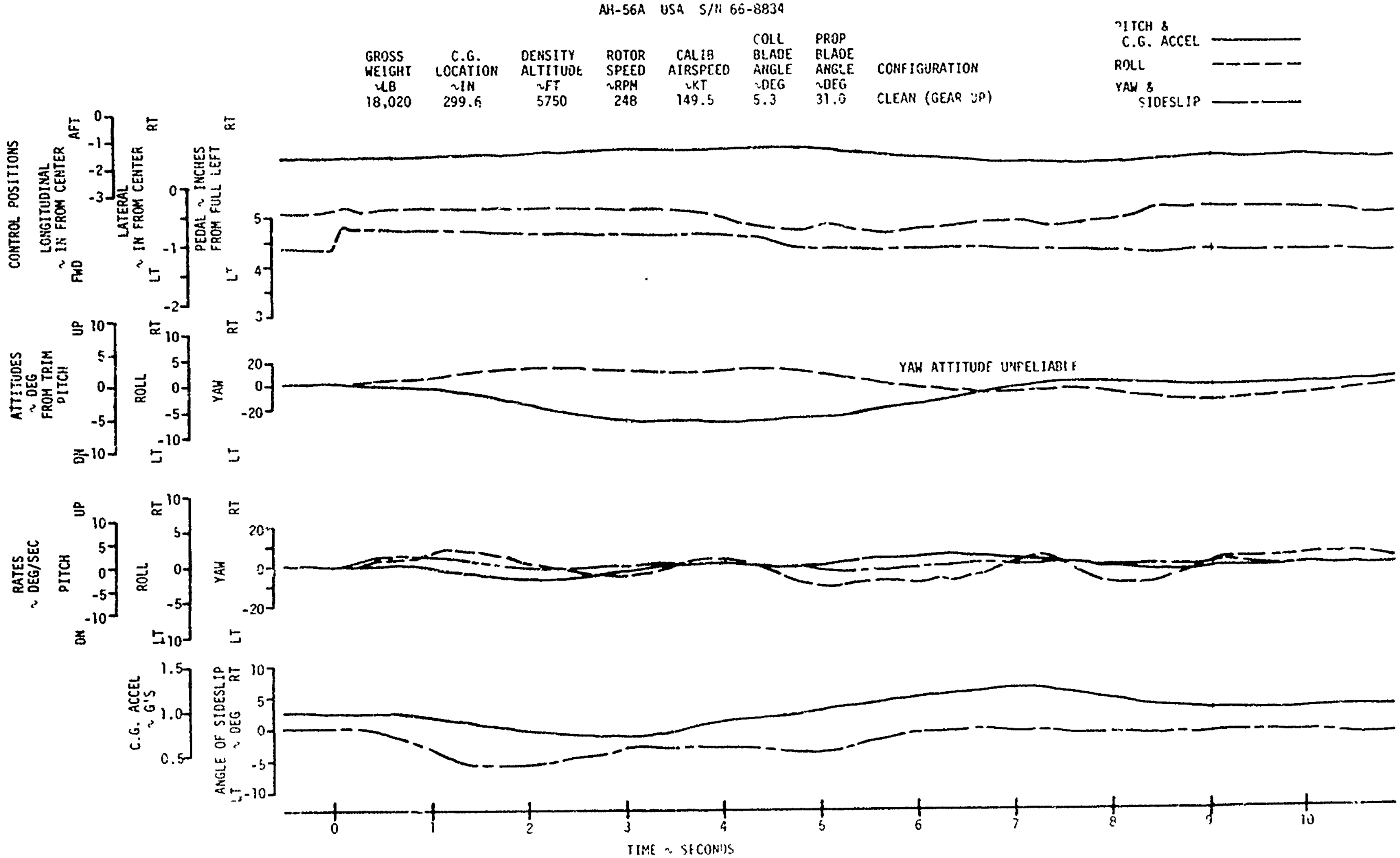


FIGURE 71

LEFT PEDAL STEP

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE GLF ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,000	299.6	5950	218	151 5	5 4	30.9	CLEAN (GEAR UP)

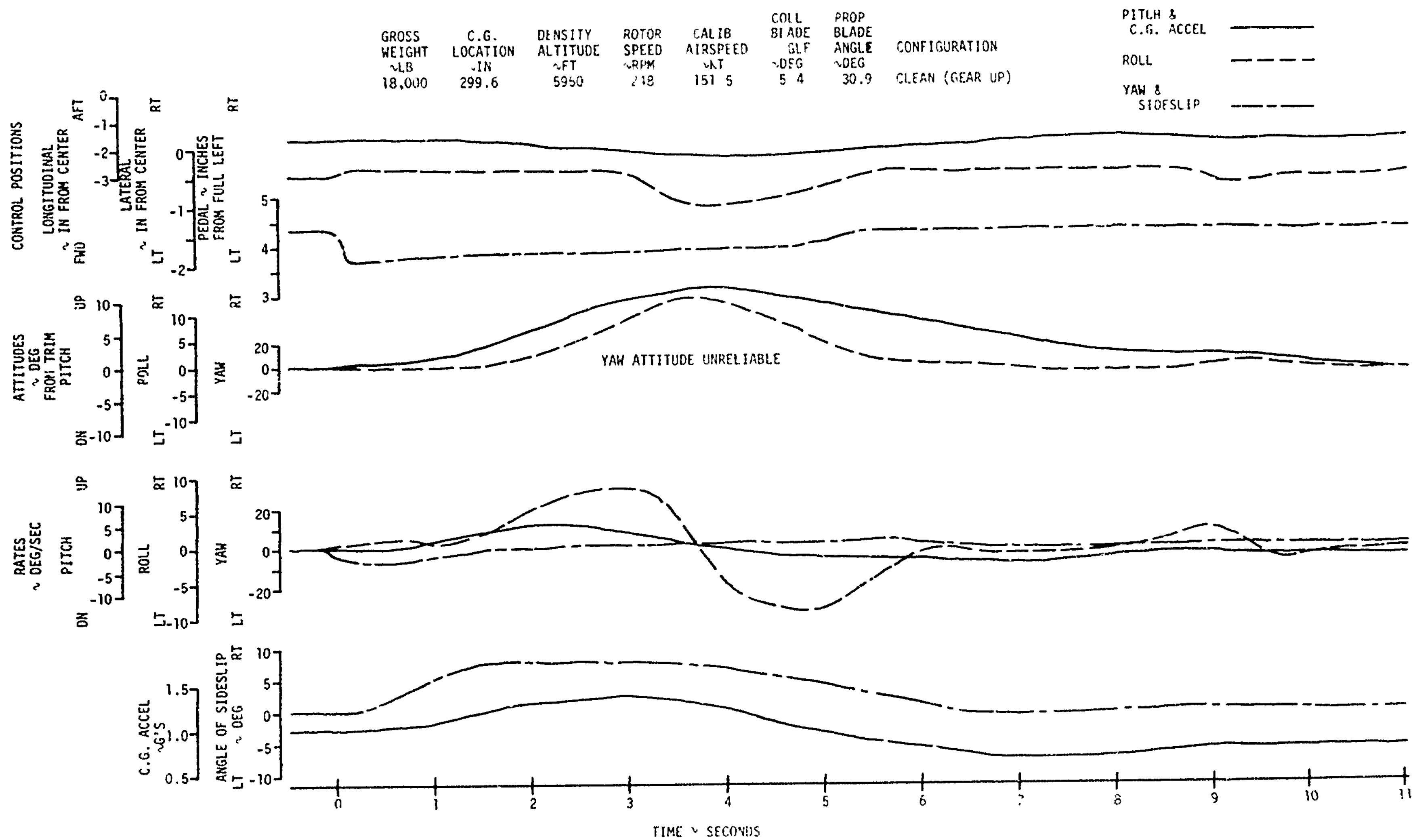


FIGURE 72

RAPID COLLECTIVE CHANGE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	PROP BLADE ANGLE ~DEG	CONFIGURATION
17,870	298.7	4360	247	62 2	13 2	CLEAN (GEAR ON)

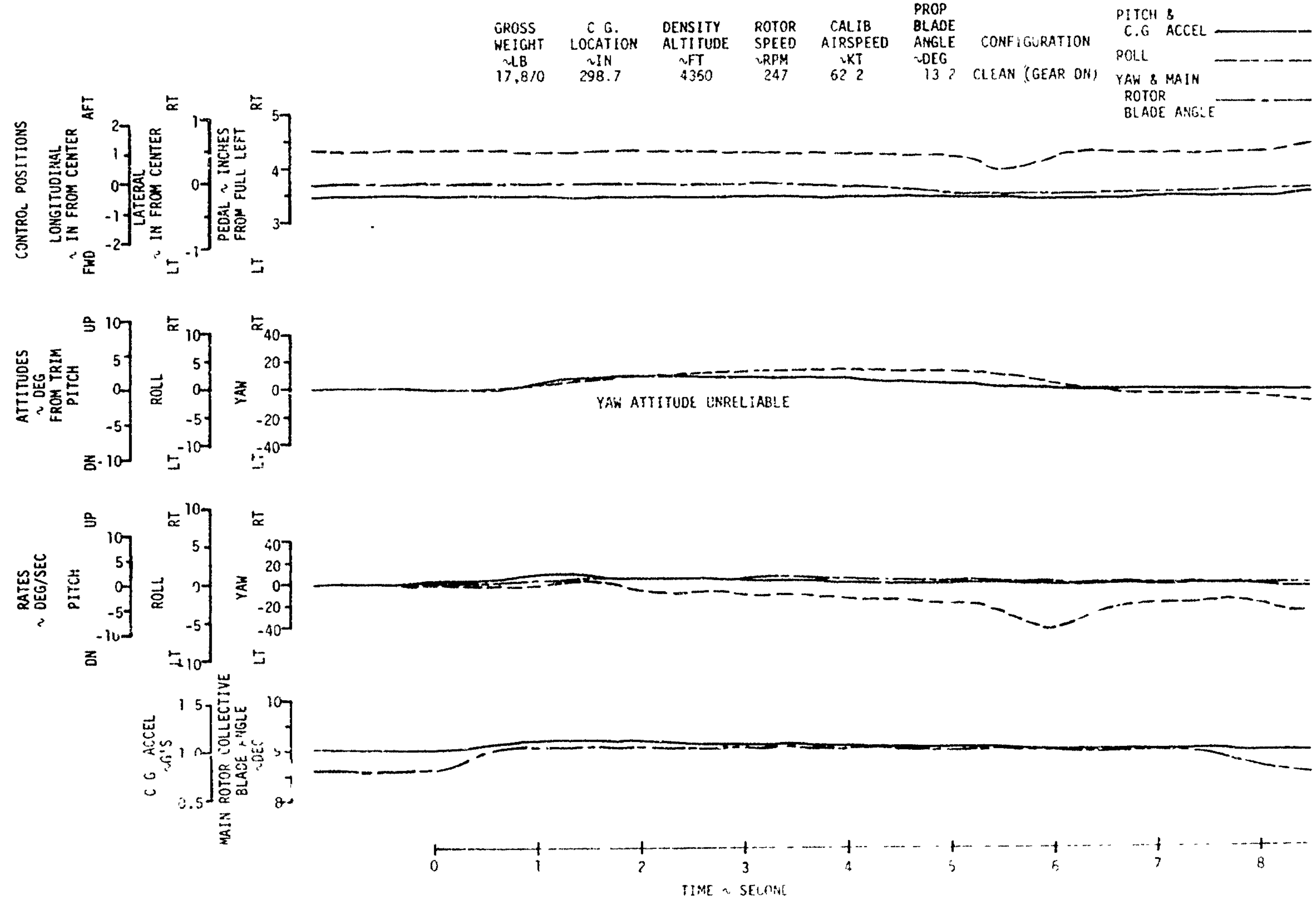
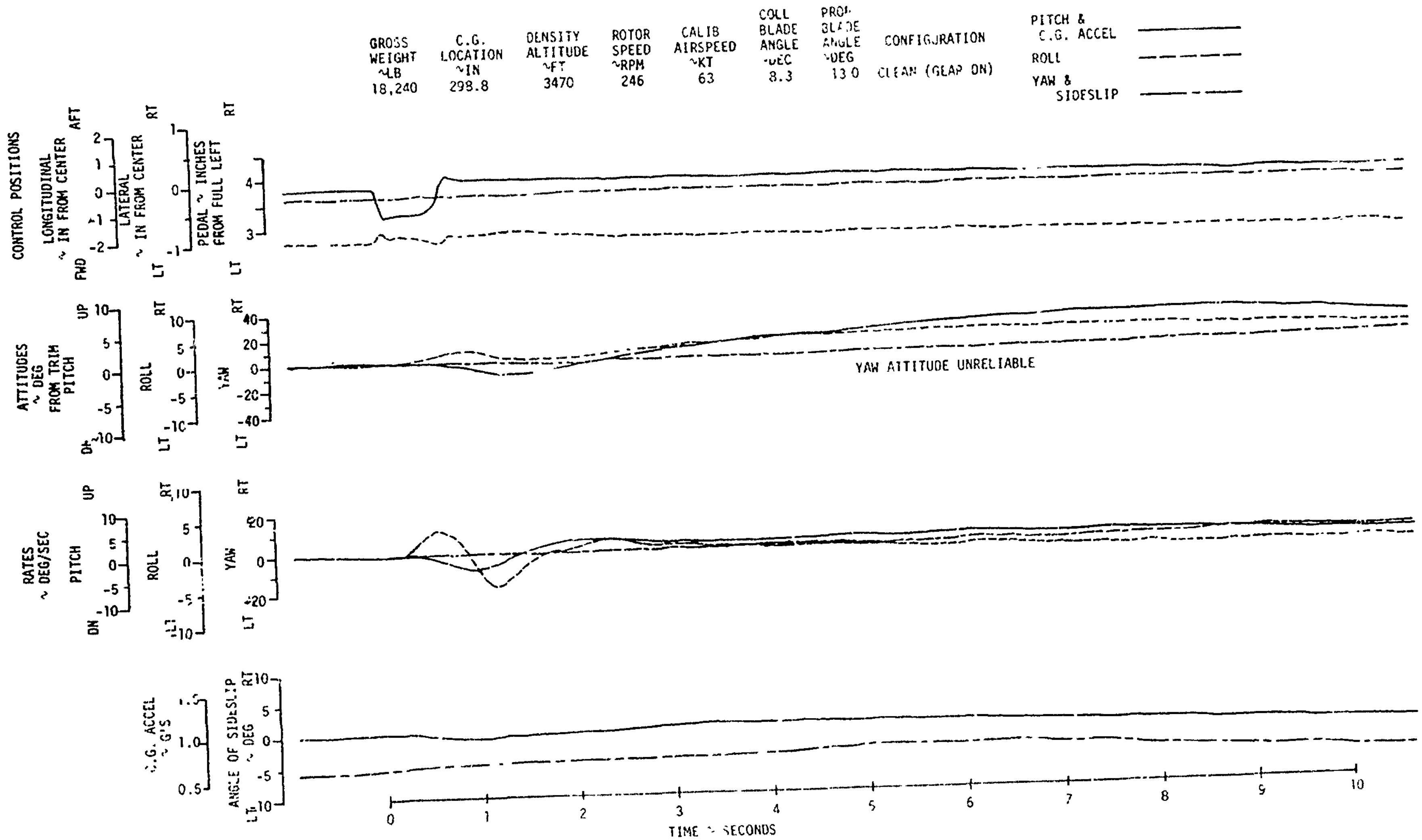


FIGURE 73

FORWARD LONGITUDINAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,240	298.8	3470	246	63	8.3	13.0	CLEAN (GLAP ON)



A.P.E. 1.1

FIGURE 74

FORWARD LONGITUDINAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,530	299.8	4100	246	101	7.0	19.6	CLEAN (GEAR UP)

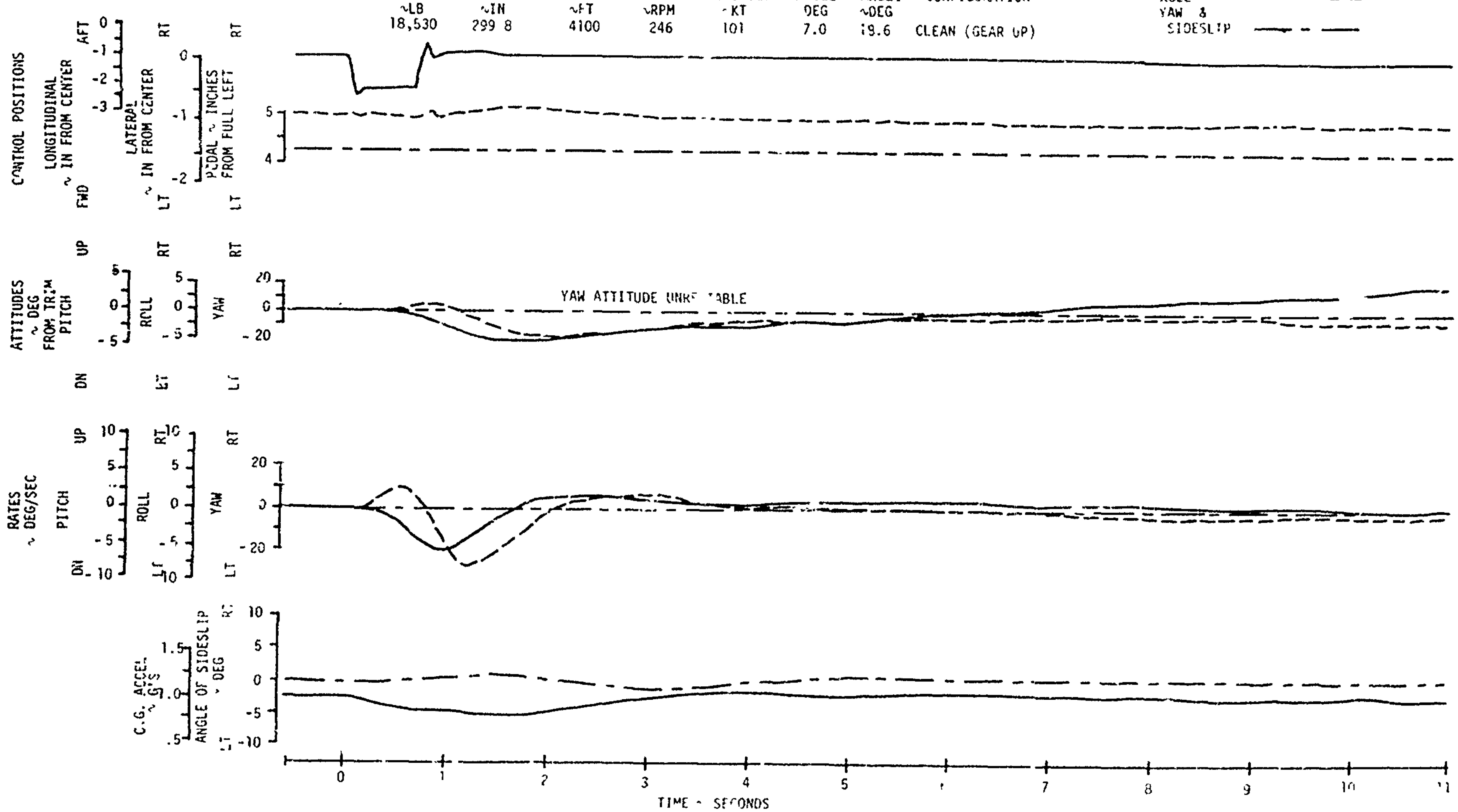


FIGURE 75

AFT LONGITUDINAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,190	299.7	4720	248	143	5.6	29.6	CLEAN (GEAR UP)

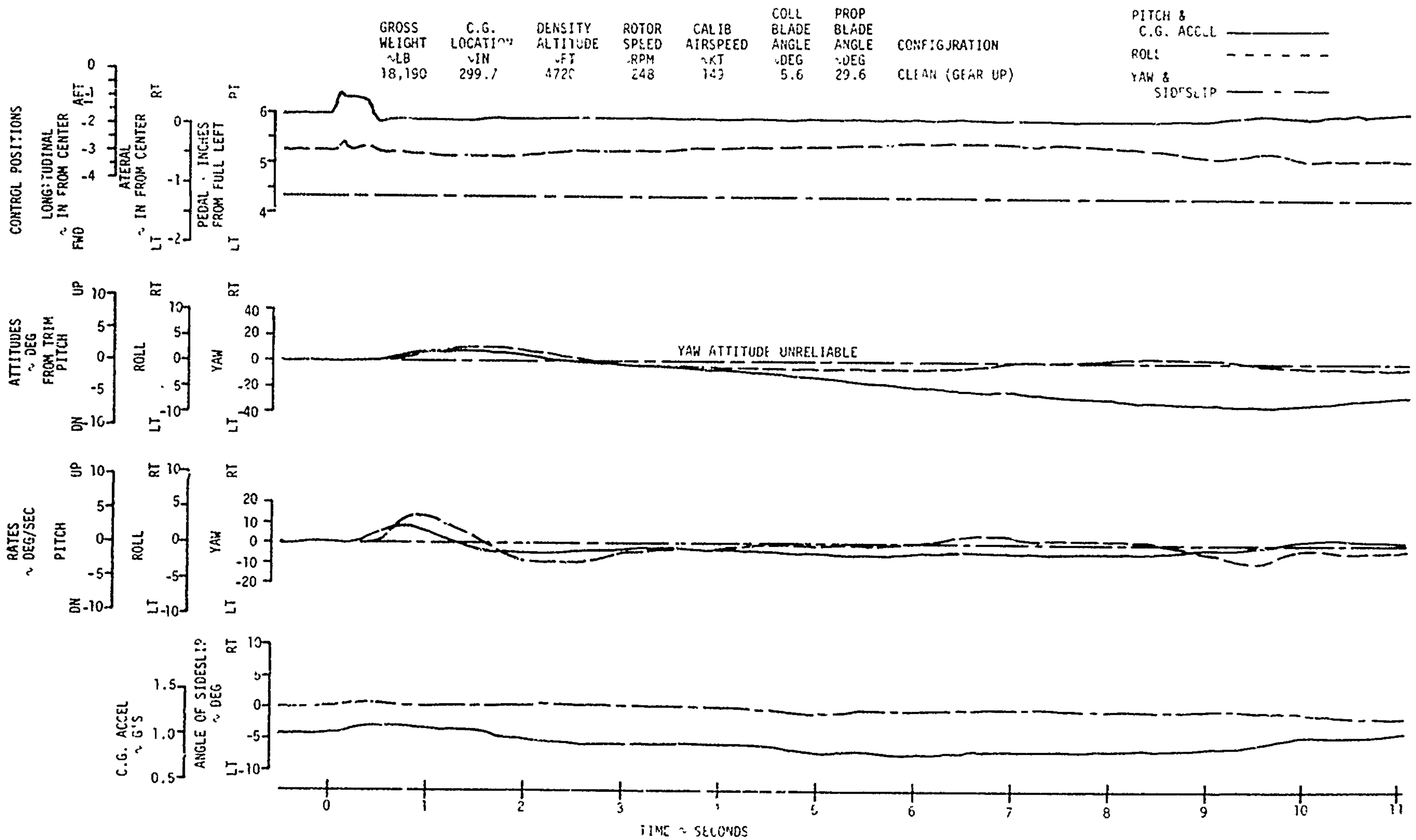


FIGURE 76

RIGHT LATERAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,360	298.8	-500	246	HOVER	12.1	-1.4	CLEAN (GEAR ON)

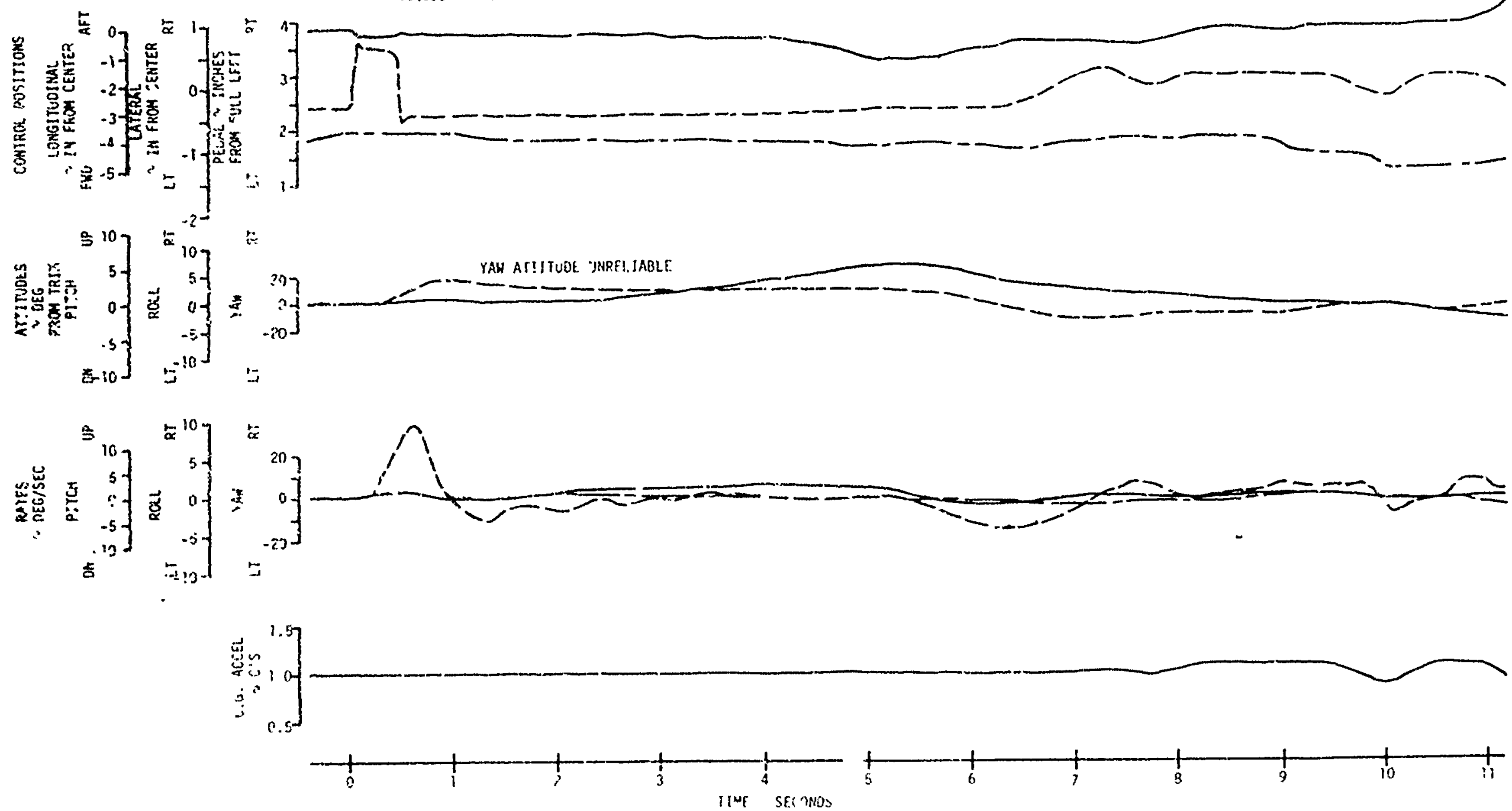


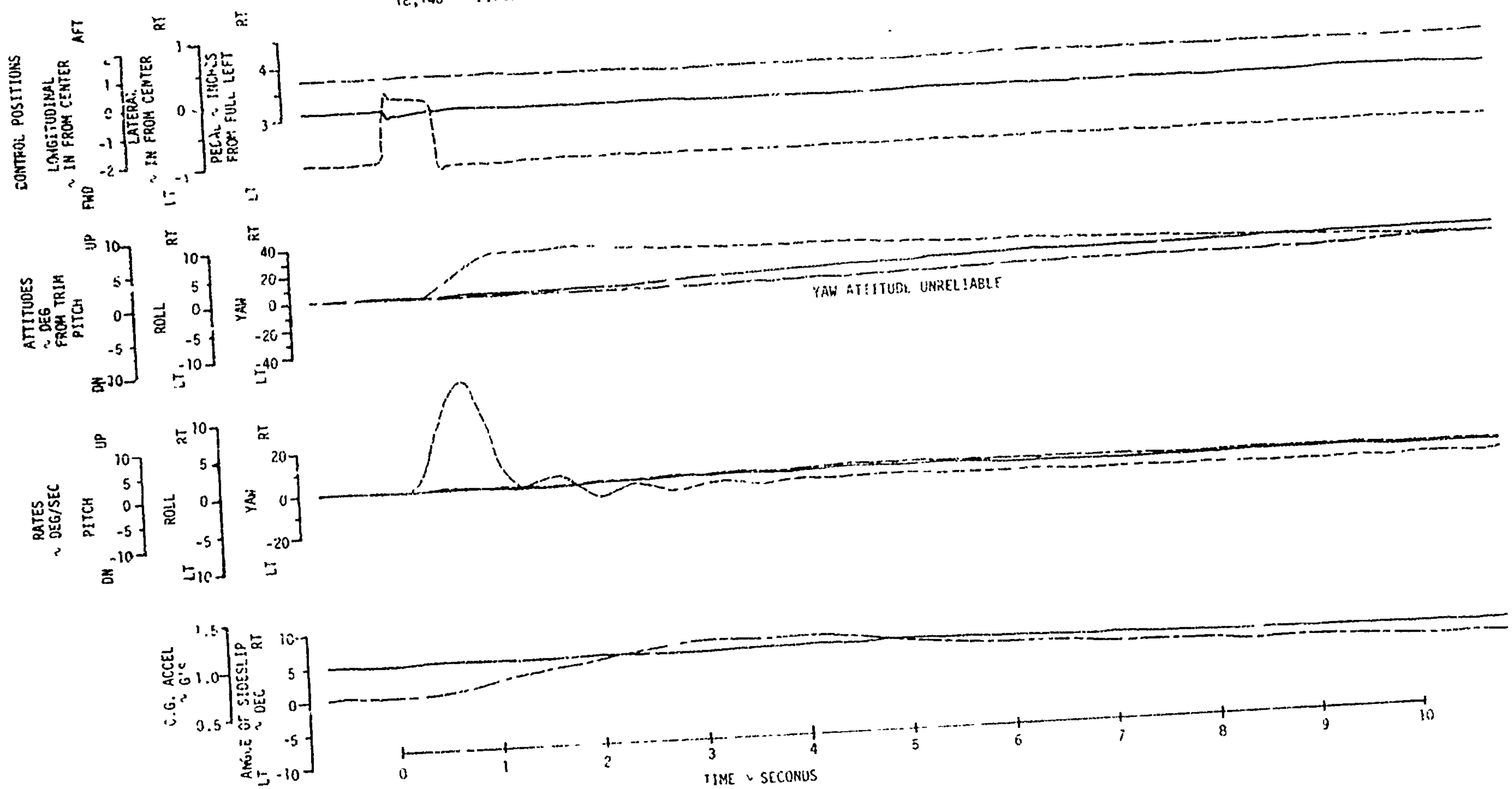
FIGURE 77

RIGHT LATERAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,140	298.7	3000	248	64	8.3	13.1	CLEAN (GEAR ON)

PITCH & C.G. ACCEL ———
ROLL - - - - -
YAW & SIDESLIP — - — -



I.URE 79
RIGHT LATERAL PULSE
AH 56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,310	299.7	3900	246	101	7 0	18 6	CLEAN (GEAR UP)

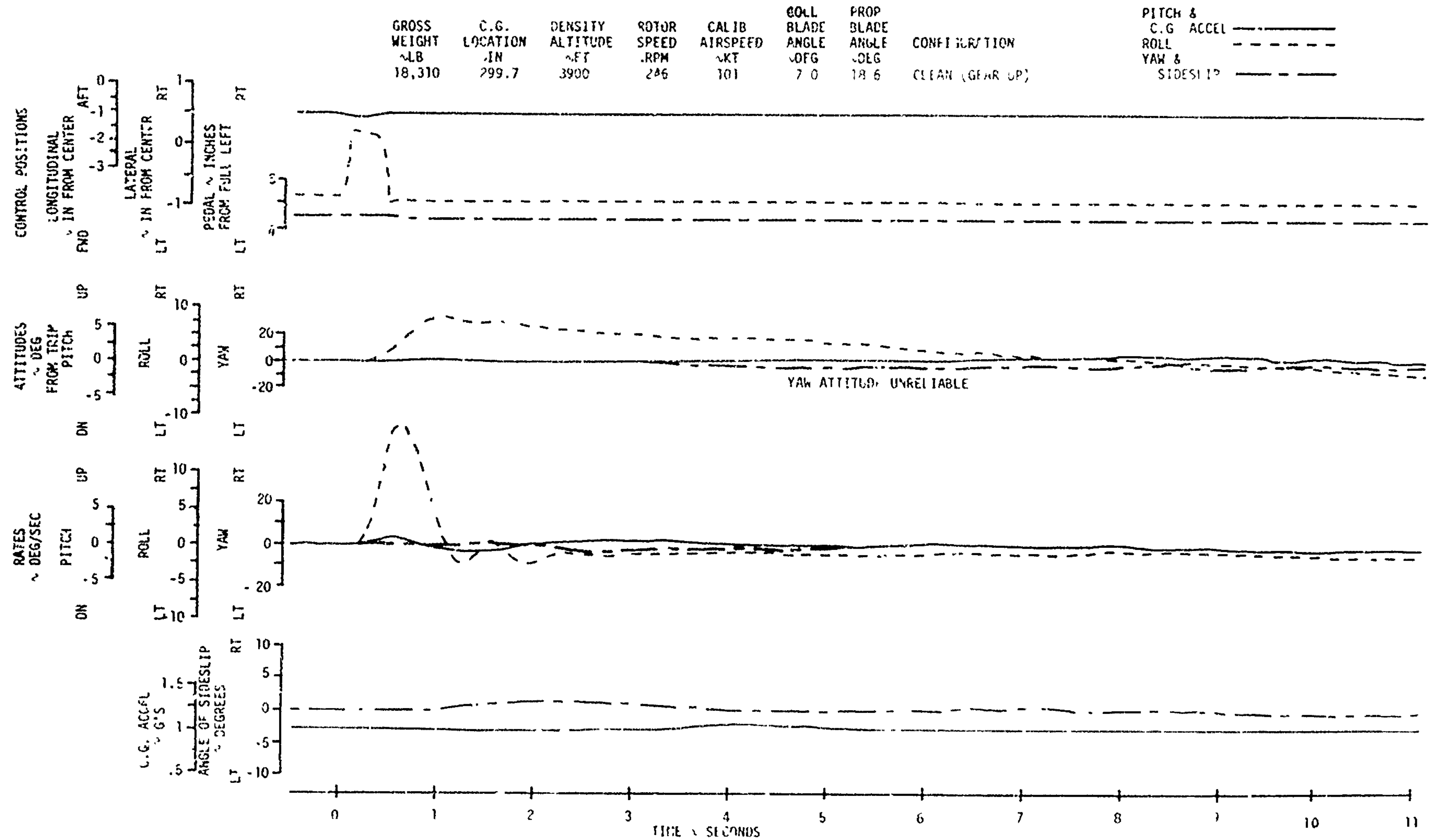


FIGURE 79

RIGHT LATERAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
17,640	29? 6	4700	248	148.5	5 6	29 6	CLEAN , GEAR UP)

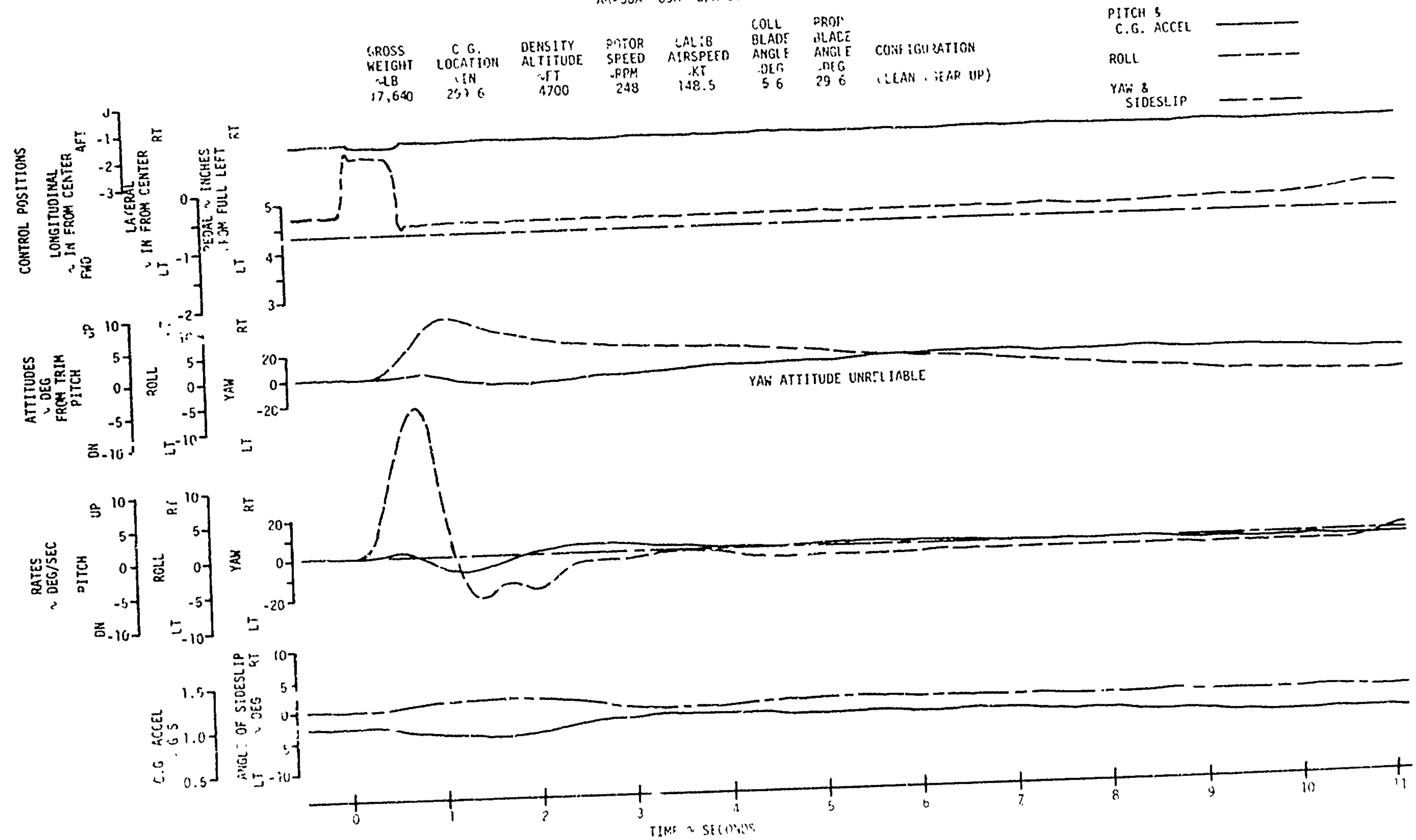


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FIGURE 8C

RIGHT LATERAL PULSE

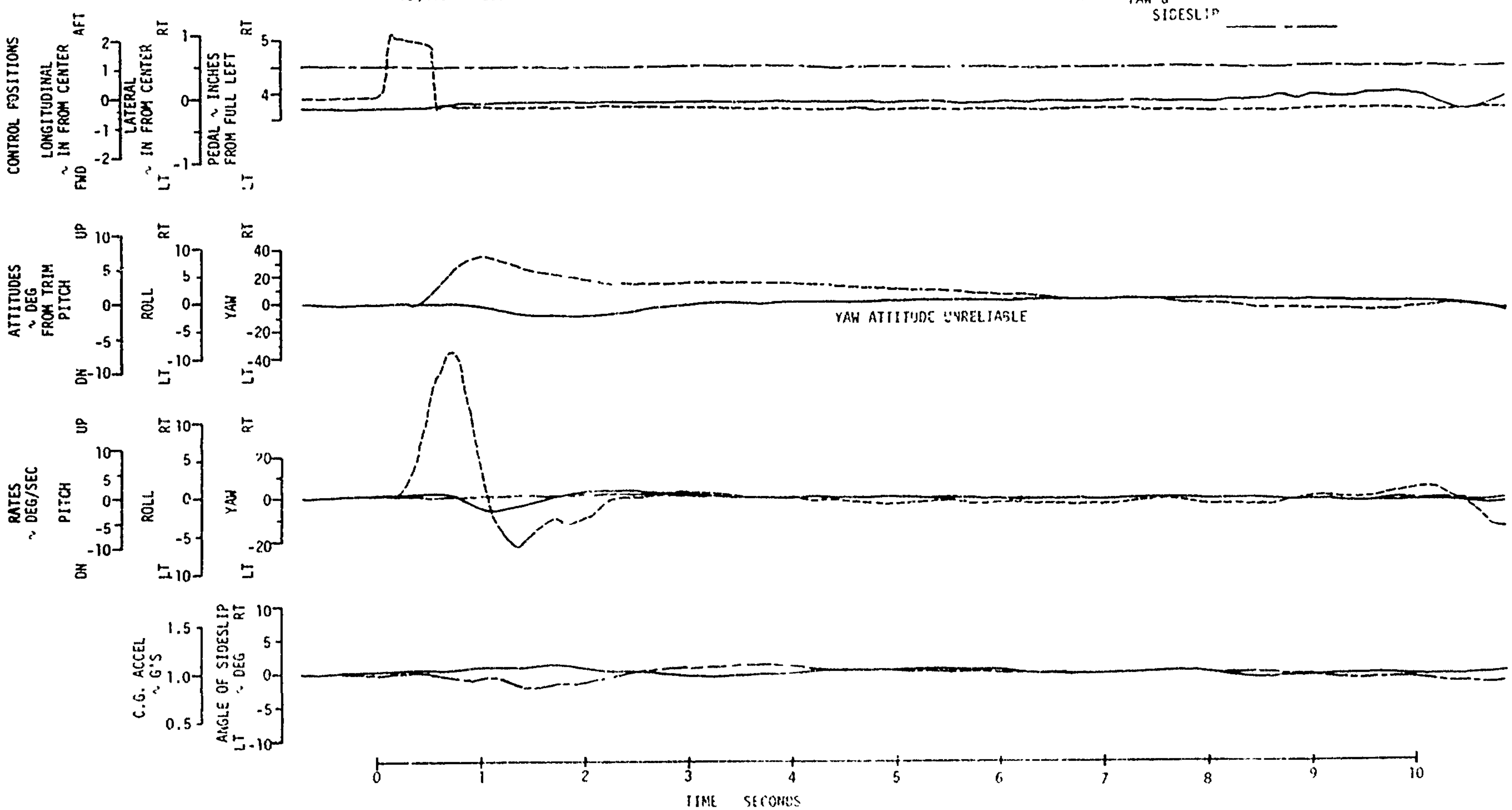
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,410	299.7	3340	247	179	5.1	32 6	CLEAN (GEAR UP)

PITCH & C.G. ACCEL ————

ROLL – – – –

YAW & SIDESLIP —— – ——



A.P.E.

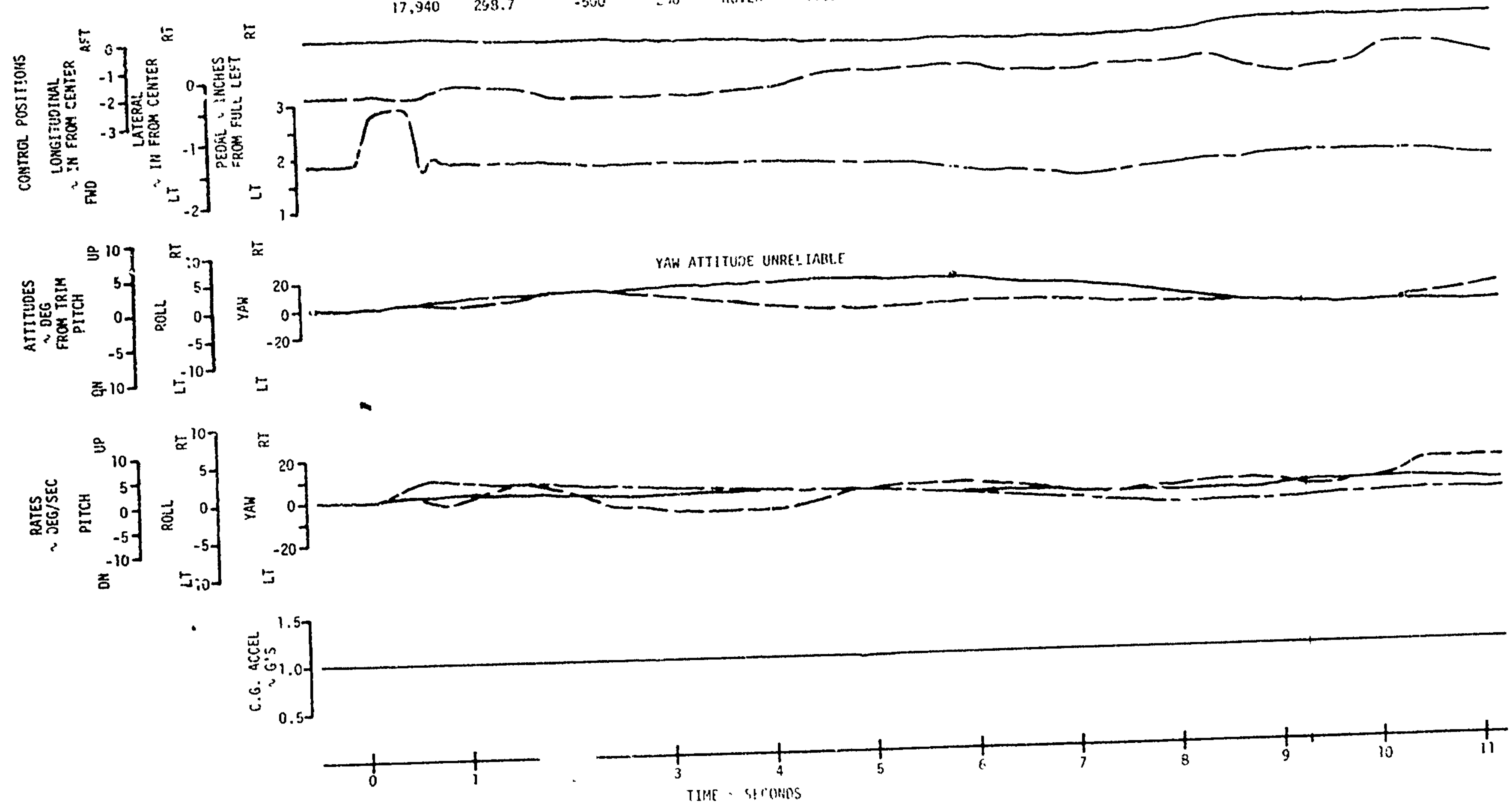
FIGURE 61

RIGHT PEDAL PULSE

AH-56A USA S/N 66 8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
17,940	298.7	-500	250	HOVER	11.7	-1.1	CLEAN (GEAR ON)

PITCH & C.G. ACCEL ————
ROLL – – – –
YAW —— – ——



A.P.L. I I

FIGURE 82

RIGHT PEDAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,010	298.7	4200	248	63	8.3	13.2	CLEAN (GEAR DN)

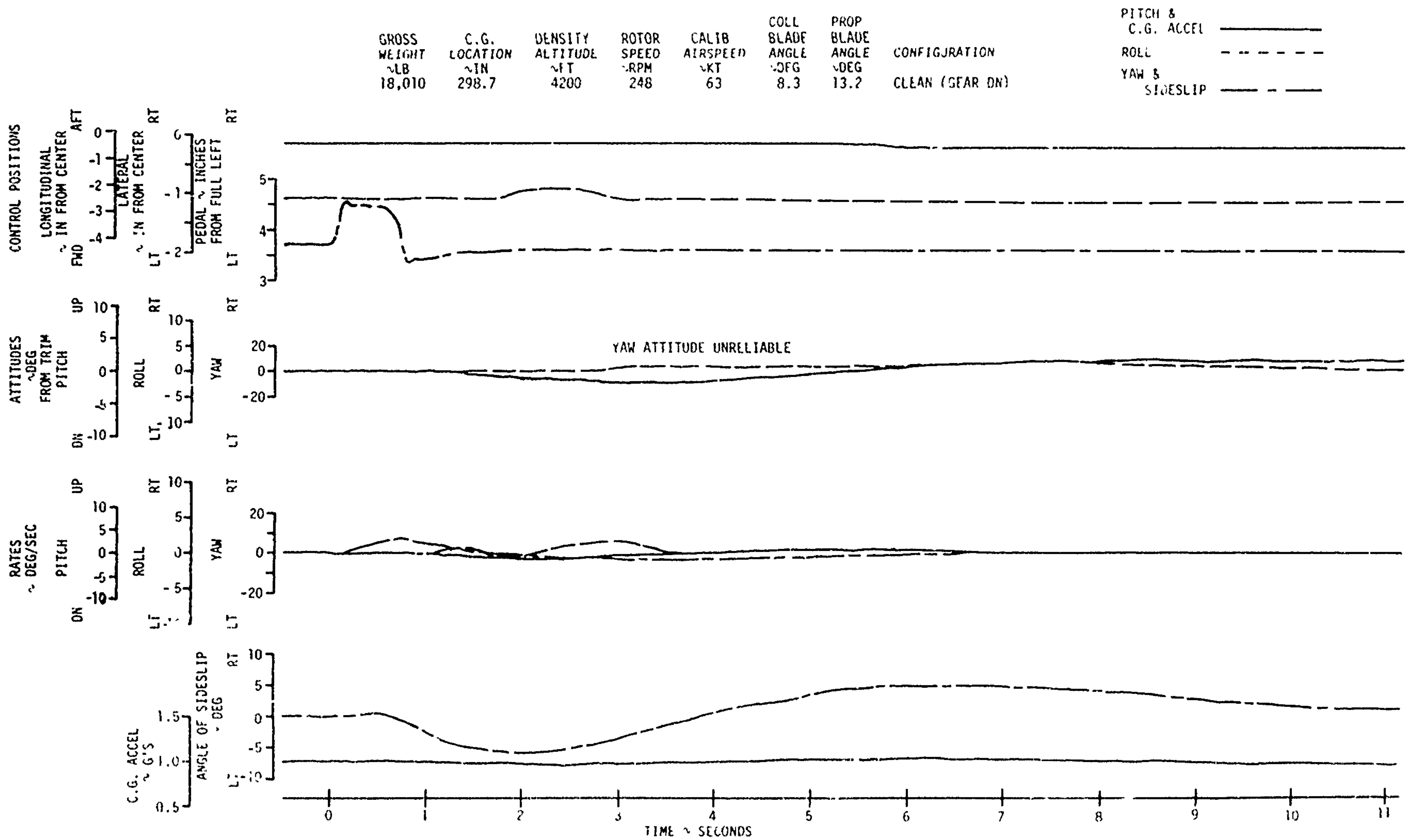


FIGURE 83

RIGHT PEDAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,170	299.6	[illegible]	246	101	7.0	18.6	CLEAN (GEAR UP)

PITCH & C.G. ACCEL ———
ROLL - - - -
YAW & SIDESLIP — — —

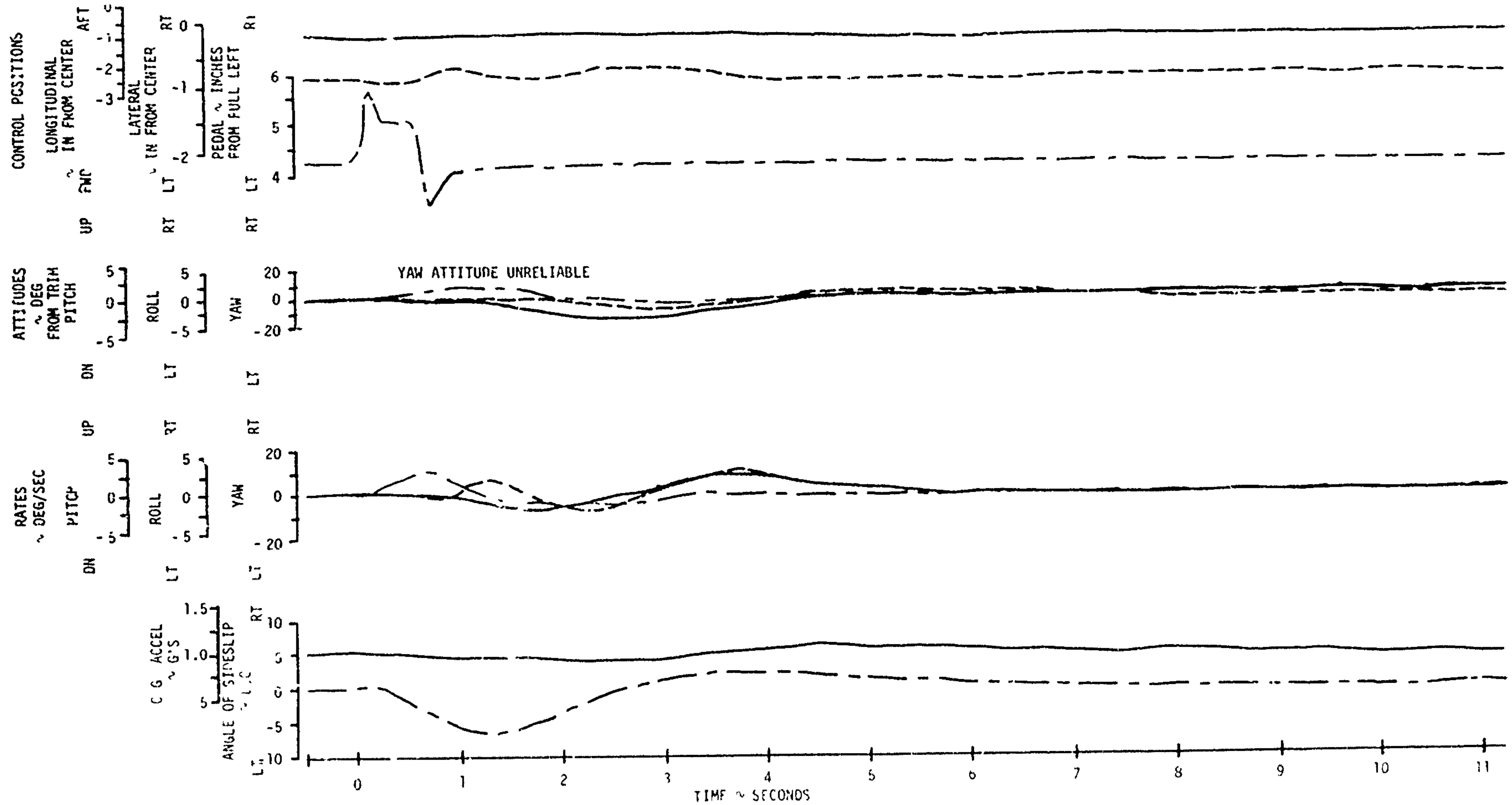


FIGURE 84
RIGHT PEDAL PULSE
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	CONFIGURATION
18,020	299 6	5900	249	150	5.4	30.9	CLEAN (GEAR UP)

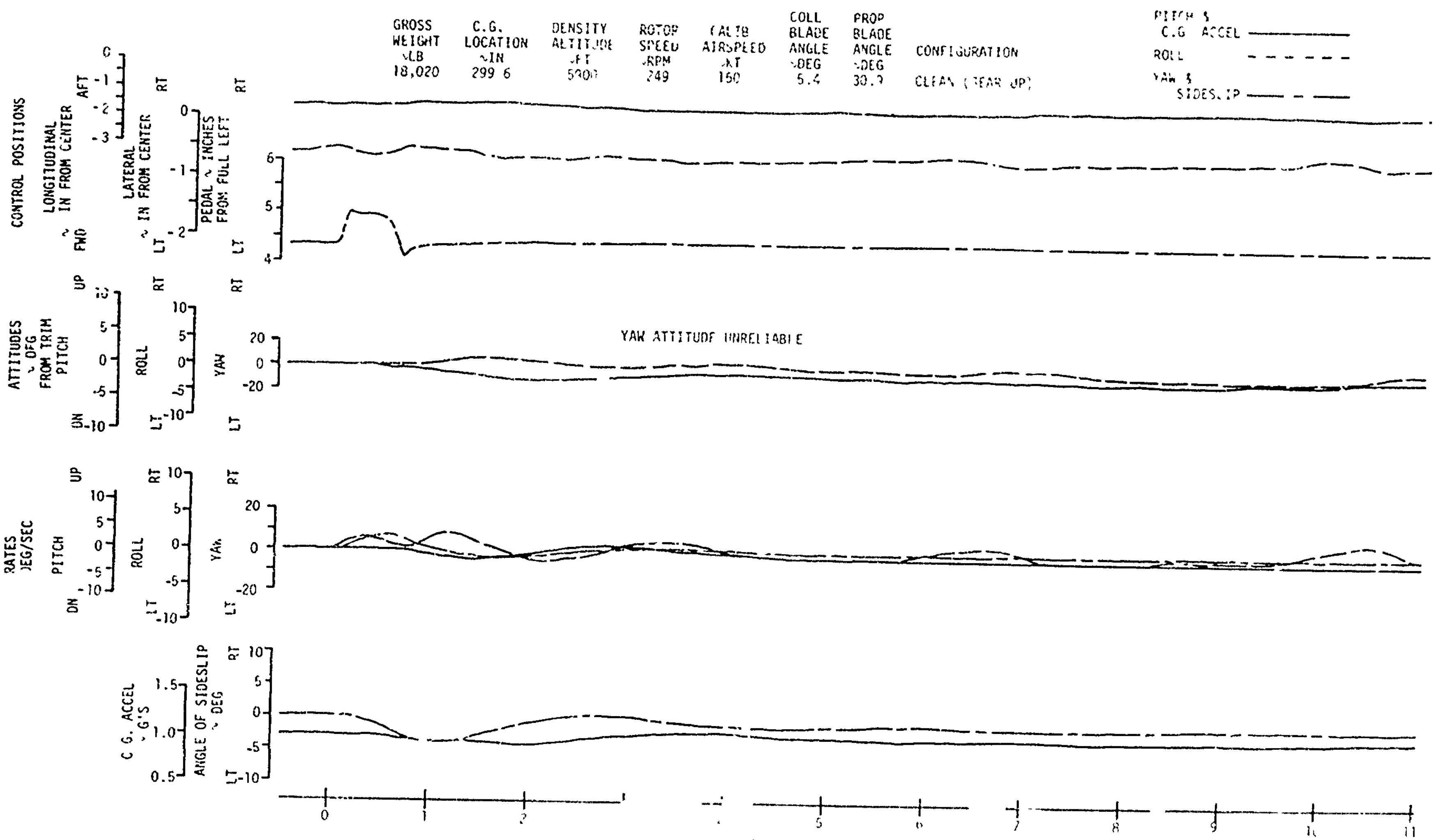


FIGURE 35

TAKE OFF FROM A HOVER

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C G LOCATION ~IN	ROTOR SPEED ~RPM	CONFIGURATION
18,680	299 8	247	CLEAN (GL"> DN)

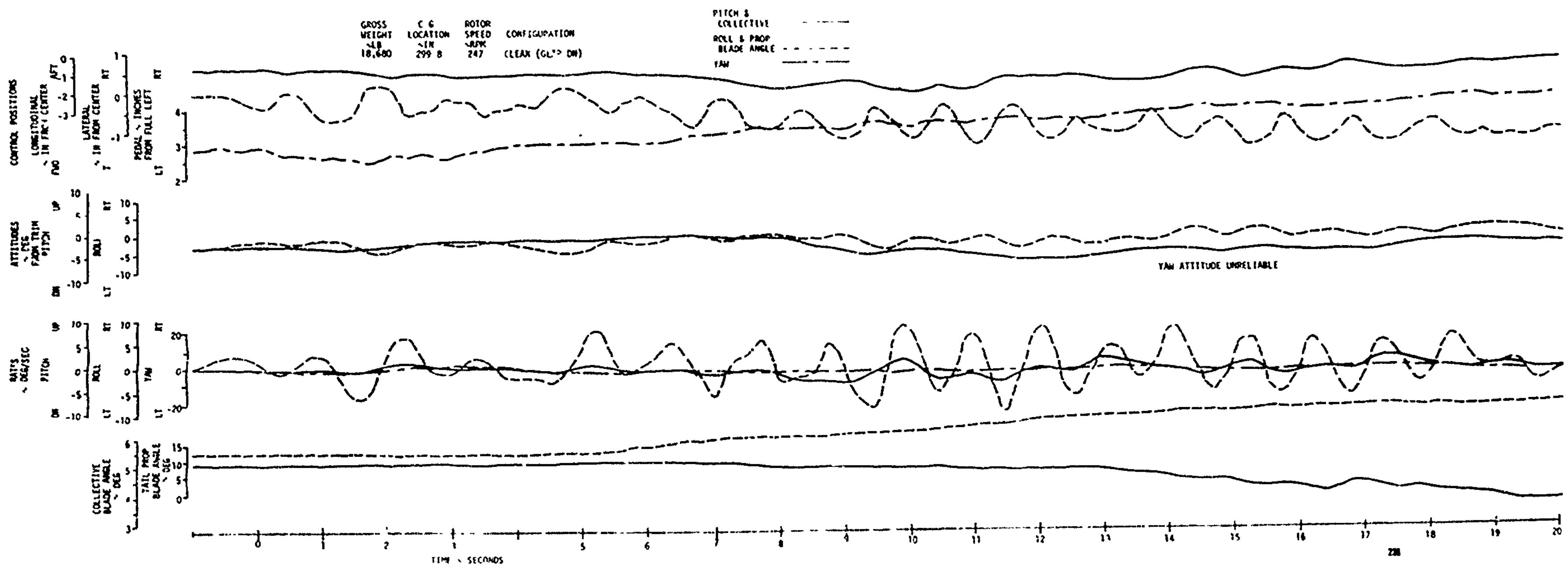


FIGURE 86

APPROACH TO HOVER

AH-56A USA S/N 66-8834

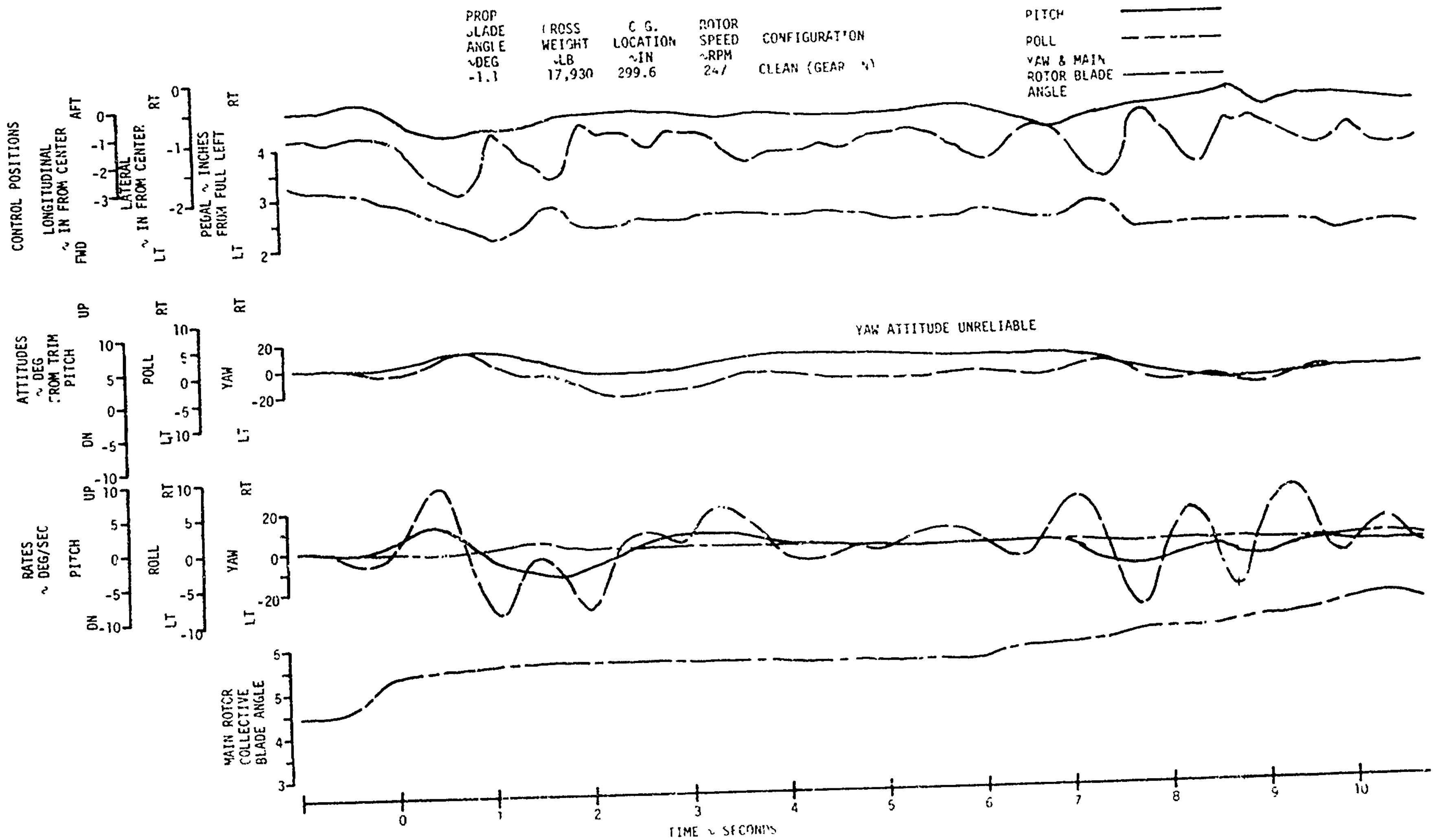
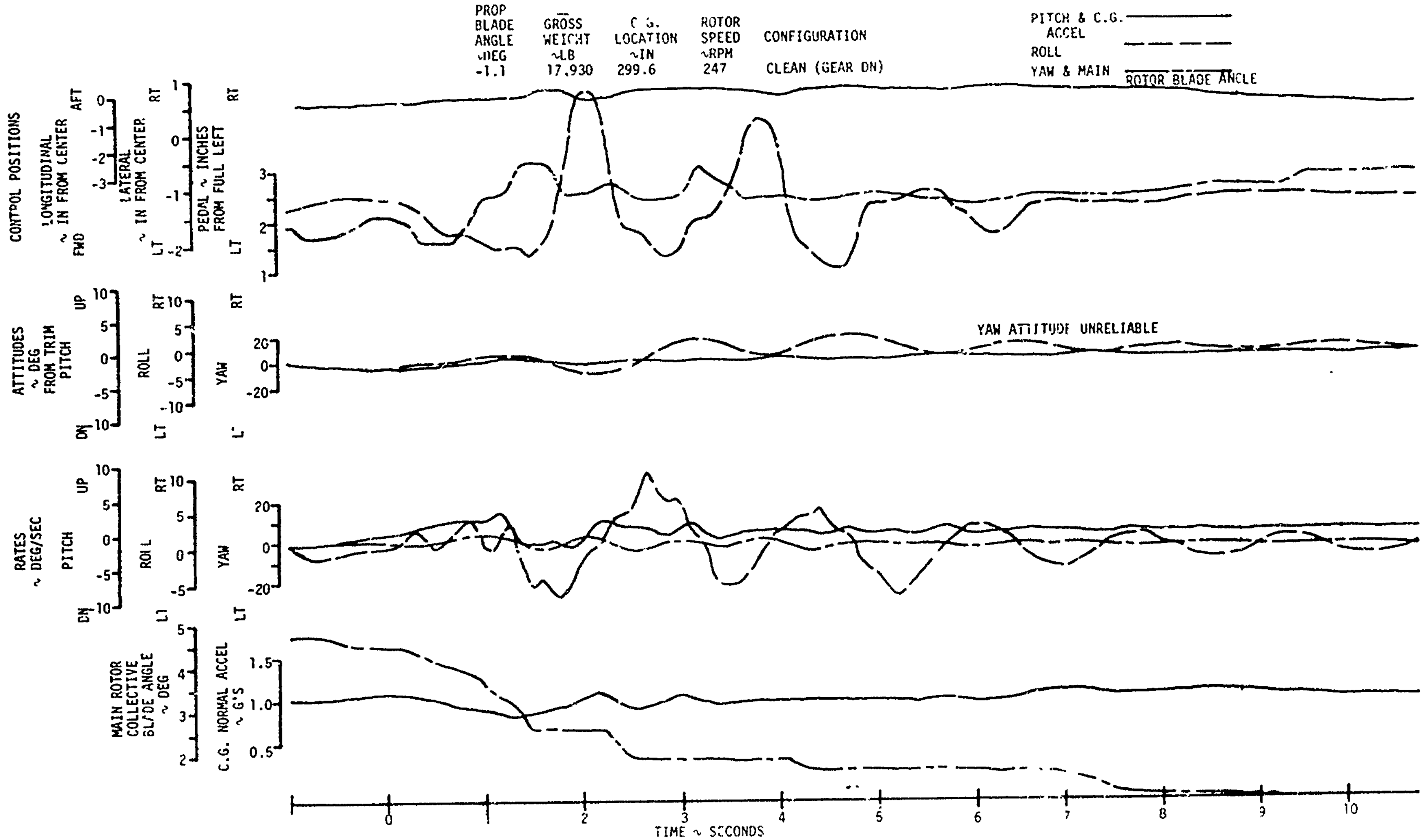


FIGURE 87

LANDING FROM HOVER

AH-56A USA S/N 66-8834



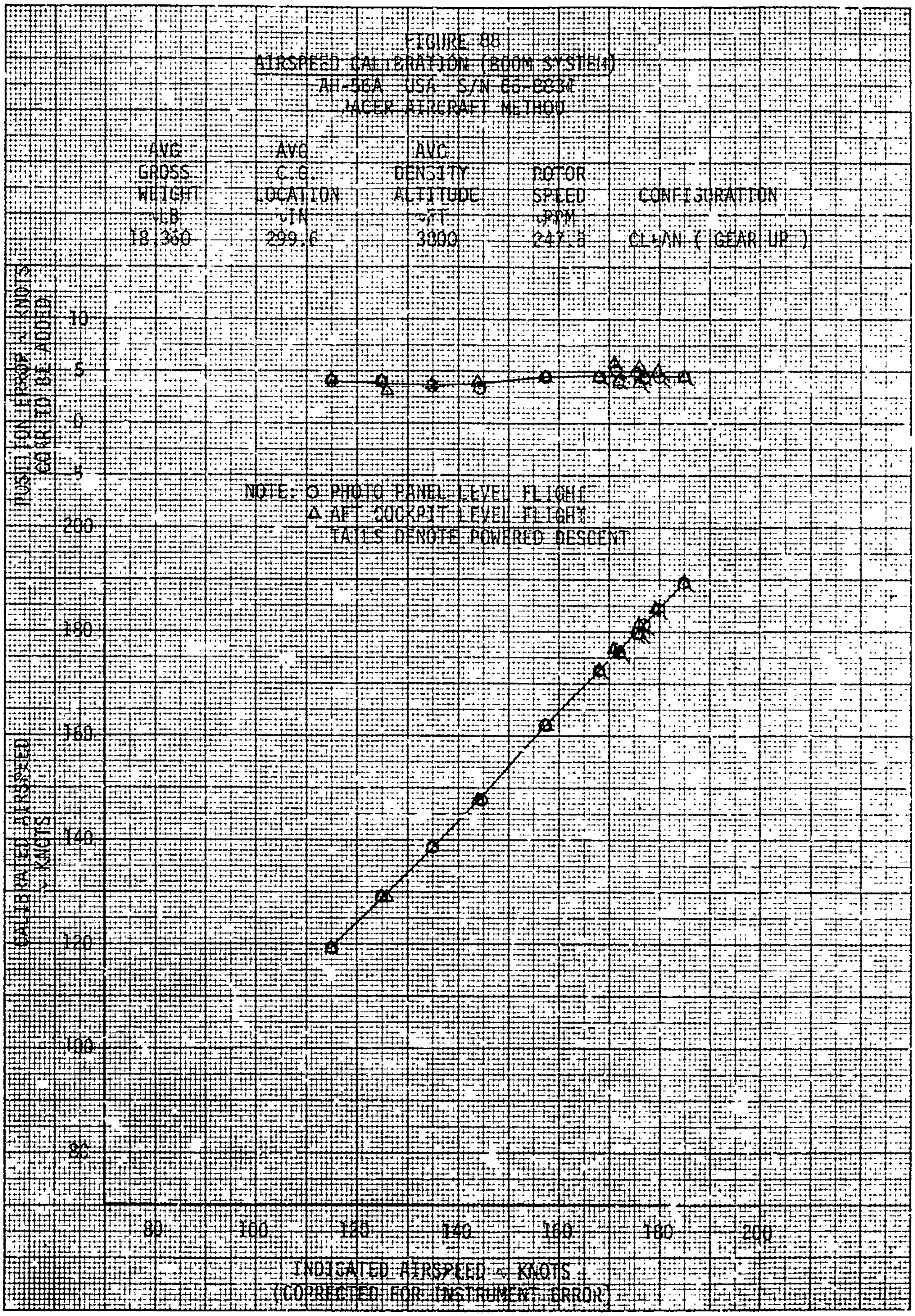


FIGURE 88
AIRSPEED CALIBRATION (BOOM SYSTEM)
AH-56A USA S/N 66-8834
PACER AIRCRAFT METHOD
AVG GROSS WEIGHT ~LB
18,360
AVG C.G. LOCATION ~IN
299.6
AVG DENSITY ALTITUDE ~FT
3000
ROTOR SPEED ~RPM
247.8
CONFIGURATION
CLEAN (GEAR UP)
POSITION ERROR ~ KNOTS
CORR TO BE ADDED
CALIBRATED AIRSPEED ~ KNOTS
NOTE: ○ PHOTO PANEL LEVEL FLIGHT
△ AFT COCKPIT LEVEL FLIGHT
TAILS DENOTE POWERED DESCENT
INDICATED AIRSPEED ~ KNOTS
(CORRECTED FOR INSTRUMENT ERROR)

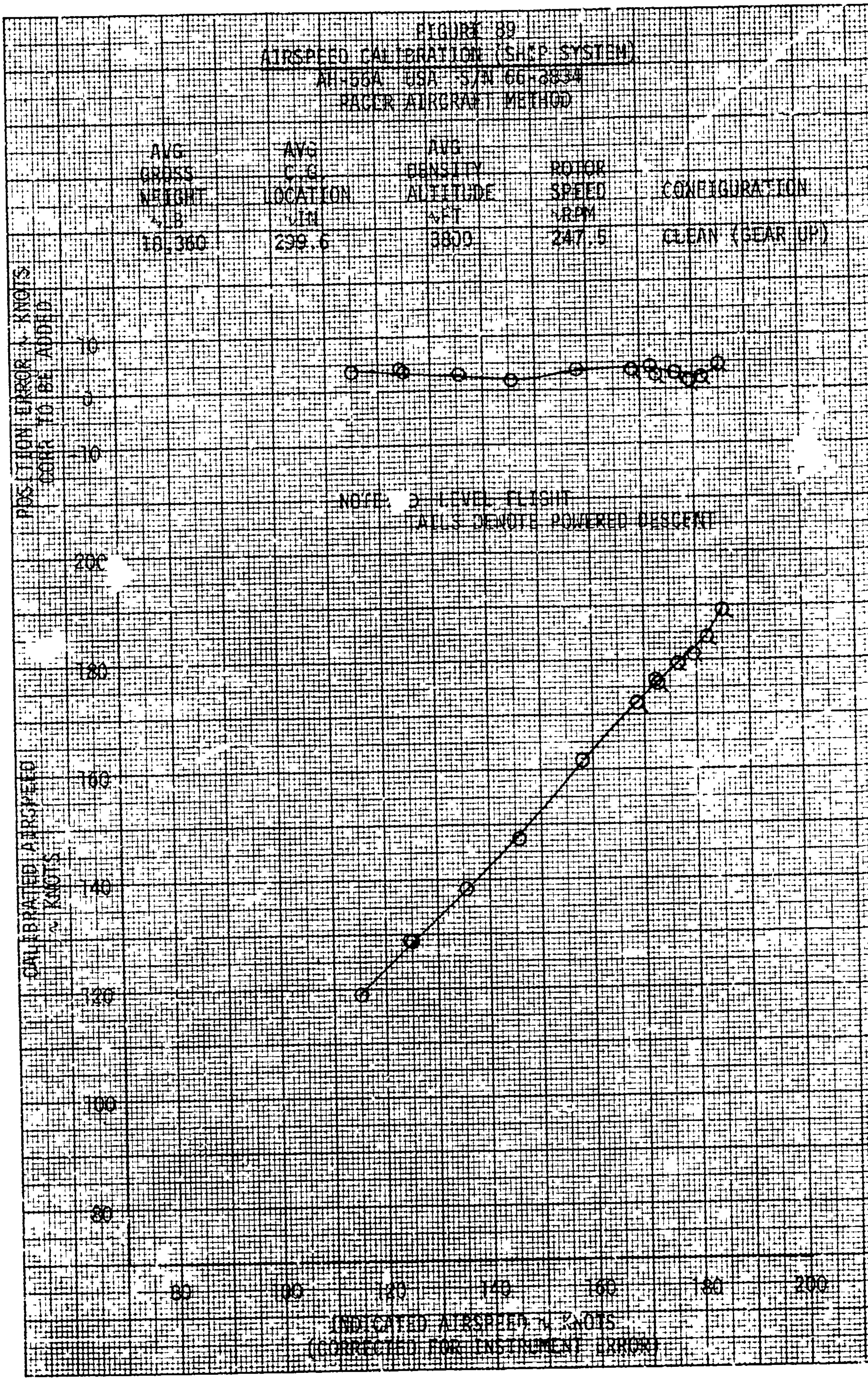
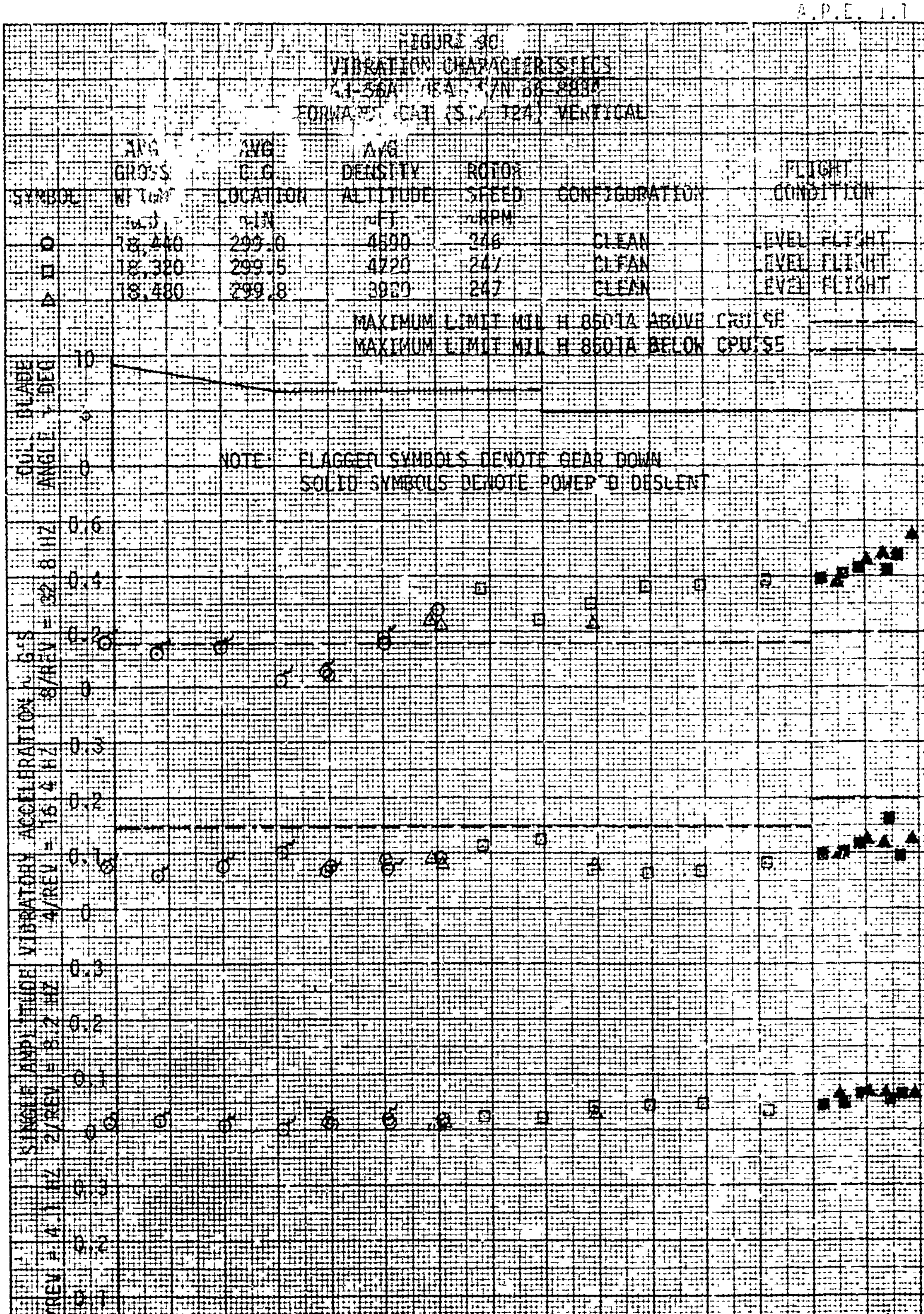


FIGURE 89
AIRSPEED CALIBRATION (SHIP SYSTEM)
AH-56A USA S/N 66-8834
PACER AIRCRAFT METHOD
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
18,360
299.6
3800
247.5
CLEAN (GEAR UP)
POSITION ERROR ~ KNOTS
CORR TO BE ADDED
10
0
-10
NOTE: LEVEL FLIGHT
TAILS DENOTE POWERED DESCENT
CALIBRATED AIRSPEED ~ KNOTS
200
180
160
140
120
100
80
80
100
120
140
160
180
200
INDICATED AIRSPEED ~ KNOTS
(CORRECTED FOR INSTRUMENT ERROR)


VIBRATION CHARACTERISTICS
VERTICAL
SYMBOL
AVG GROSS WEIGHT
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
FLIGHT CONDITION
18,440
299.0
4690
246
CLEAN
LEVEL FLIGHT
18,320
299.5
4720
247
CLEAN
LEVEL FLIGHT
18,480
299.8
3920
247
CLEAN
LEVEL FLIGHT
MAXIMUM LIMIT MIL H 8501A ABOVE CRUISE
MAXIMUM LIMIT MIL H 8501A BELOW CRUISE
NOTE: FLAGGED SYMBOLS DENOTE GEAR DOWN
SOLID SYMBOLS DENOTE POWERED DESCENT
COLL. BLADE ANGLE ~ DEG
SINGLE AMPLITUDE VIBRATORY ACCELERATION ~ G'S
1/REV = 4.1 HZ
2/REV = 8.2 HZ
4/REV = 16.4 HZ
8/REV = 32.8 HZ
CALIBRATED AIRSPEED ~ KNOTS
40
60
80
100
120
140
160

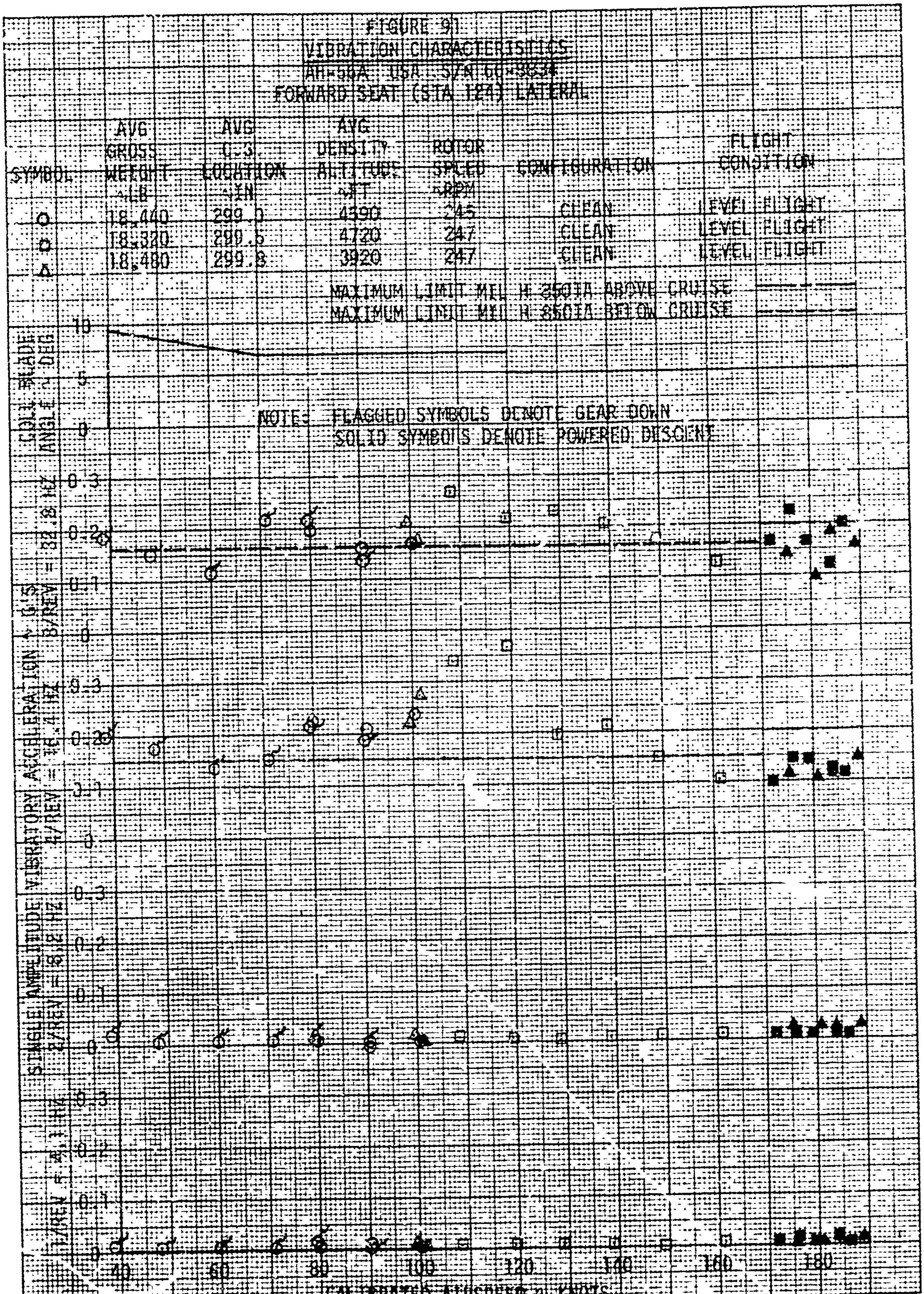


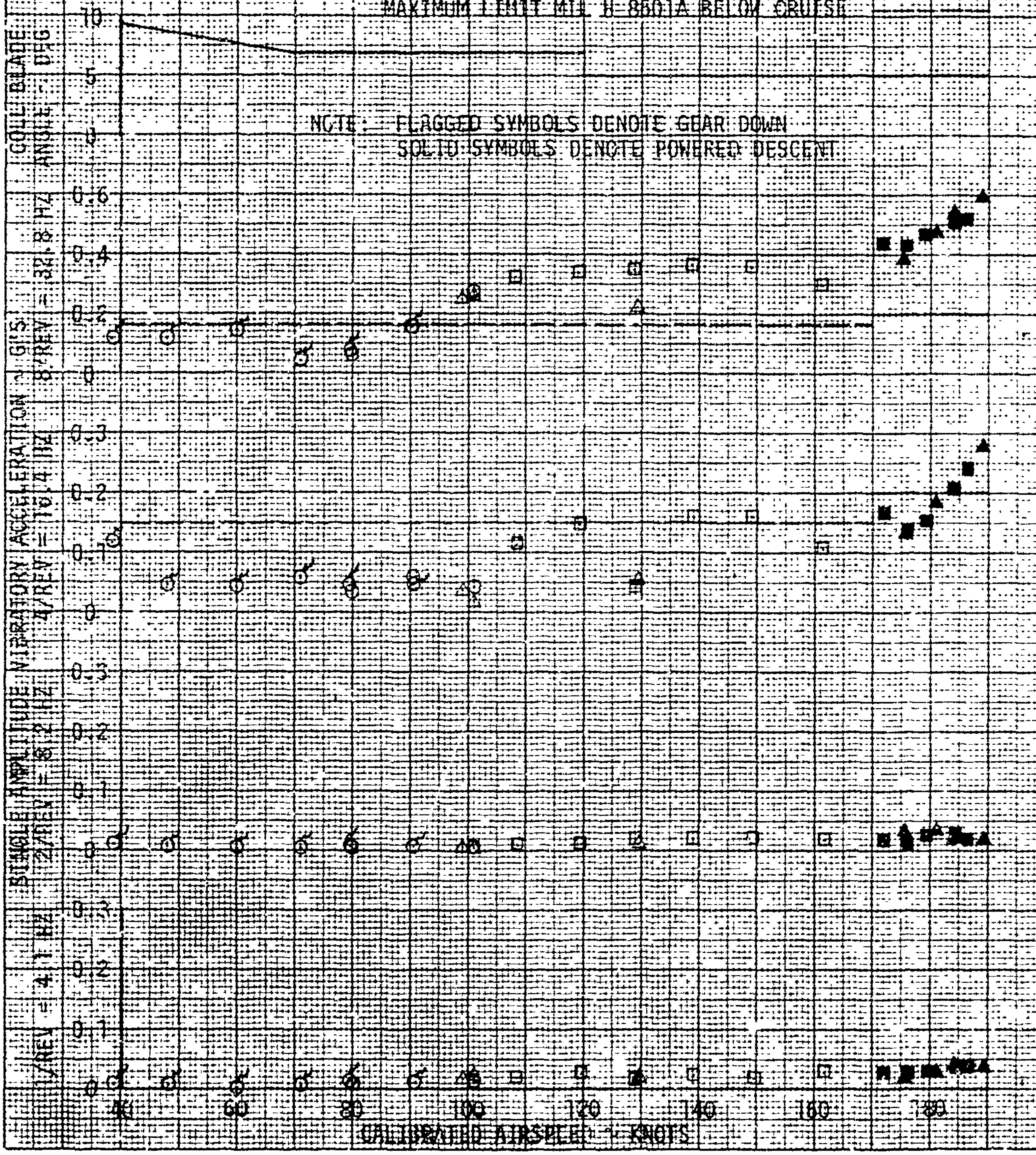
FIGURE 91
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
FORWARD SEAT (STA 124) LATERAL
SYMBOL
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
FLIGHT CONDITION
18,440
299.0
4390
245
CLEAN
LEVEL FLIGHT
18,320
290.5
4720
247
CLEAN
LEVEL FLIGHT
18,480
299.8
3920
247
CLEAN
LEVEL FLIGHT
MAXIMUM LIMIT MIL H 8501A ABOVE CRUISE
MAXIMUM LIMIT MIL H 8501A BELOW CRUISE
NOTE: FLAGGED SYMBOLS DENOTE GEAR DOWN
SOLID SYMBOLS DENOTE POWERED DESCENT
COLL BLADE ANGLE ~ DEG
SINGLE AMPLITUDE VIBRATORY ACCELERATION ~ G'S
1/REV = 4.1 HZ 2/REV = 8.2 HZ 4/REV = 16.4 HZ 8/REV = 32.8 HZ
CALIBRATED AIRSPEED ~ KNOTS
10
5
0
0.3
0.2
0.1
0
40
60
80
100
120
140
160
180

A.P.C. 1.1

FIGURE 32
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8830
AFT SEAT (STA 120) VERTICAL

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
○	18,440	299.0	4590	245	CLEAN	LEVEL FLIGHT
□	18,320	299.5	4720	247	CLEAN	LEVEL FLIGHT
△	18,480	299.8	3920	247	CLEAN	LEVEL FLIGHT

MAXIMUM LIMIT MIL H 8501A ABOVE CRUISE
MAXIMUM LIMIT MIL H 8501A BELOW CRUISE



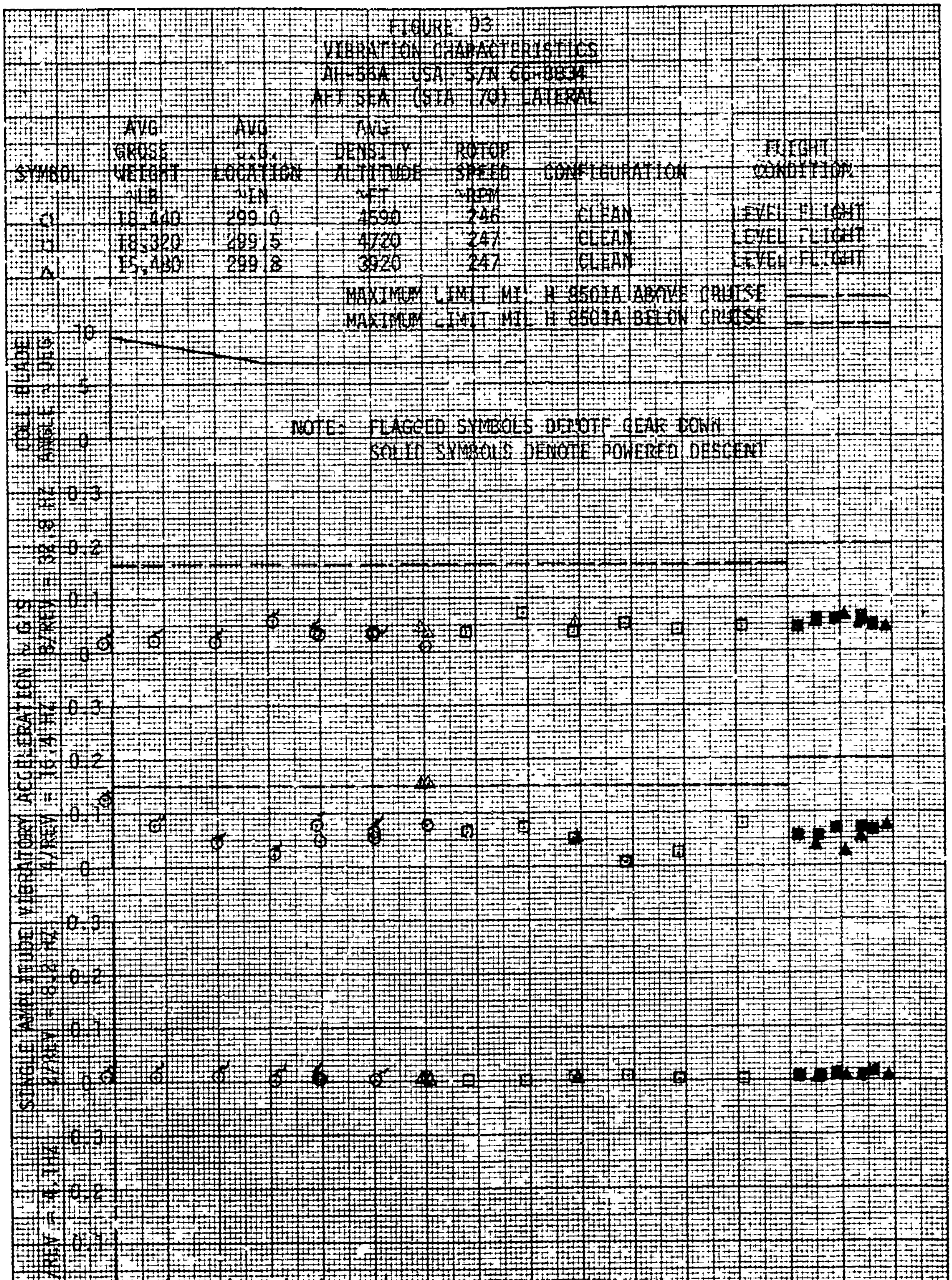


FIGURE 93
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
AFT SEAT (STA 170) LATERAL
SYMBOL
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
FLIGHT CONDITION
18,440 299.0 4590 246 CLEAN LEVEL FLIGHT
18,320 299.5 4720 247 CLEAN LEVEL FLIGHT
15,480 299.8 3920 247 CLEAN LEVEL FLIGHT
MAXIMUM LIMIT MIL H 8501A ABOVE CRUISE
MAXIMUM LIMIT MIL H 8501A BELOW CRUISE
NOTE: FLAGGED SYMBOLS DENOTE GEAR DOWN
SOLID SYMBOLS DENOTE POWERED DESCENT
SINGLE AMPLITUDE VIBRATORY ACCELERATION ~ G'S
COLL BLADE ANGLE ~ DEG
1/REV = 4.1HZ
2/REV = 8.2 HZ
4/REV = 16.4 HZ
8/REV = 32.8 HZ
CALIBRATED AIRSPEED ~ KNOTS
40
60
80
100
120
140
160
180

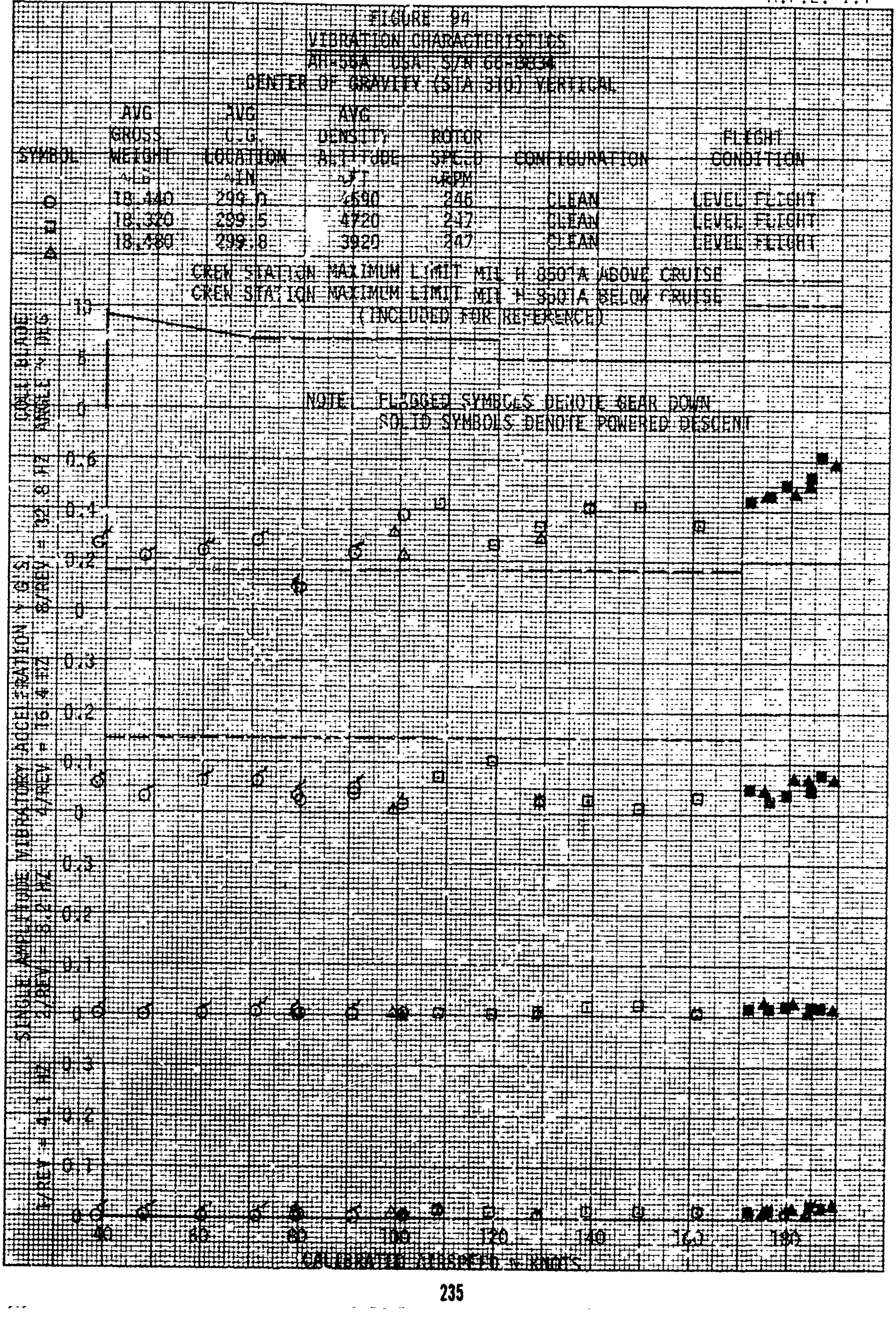


FIGURE 94
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
CENTER OF GRAVITY (STA 310) VERTICAL
SYMBOL
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
CONFIGURATION
FLIGHT CONDITION
18,440 299.0 4690 246 CLEAN LEVEL FLIGHT
18,320 299.5 4720 247 CLEAN LEVEL FLIGHT
18,480 299.8 3920 247 CLEAN LEVEL FLIGHT
CREW STATION MAXIMUM LIMIT MIL H 8501A ABOVE CRUISE
CREW STATION MAXIMUM LIMIT MIL H 8501A BELOW CRUISE
(INCLUDED FOR REFERENCE)
NOTE: FLAGGED SYMBOLS DENOTE GEAR DOWN
SOLID SYMBOLS DENOTE POWERED DESCENT
COLL BLADE ANGLE ~ DEG
SINGLE AMPLITUDE VIBRATORY ACCELERATION ~ G'S
1/REV = 4.1 HZ 2/REV = 8.2 HZ 4/REV = 16.4 HZ 8/REV = 32.8 HZ
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 96
LONGITUDINAL STICK FORCES
AH-1RA USA S/N 66-8834

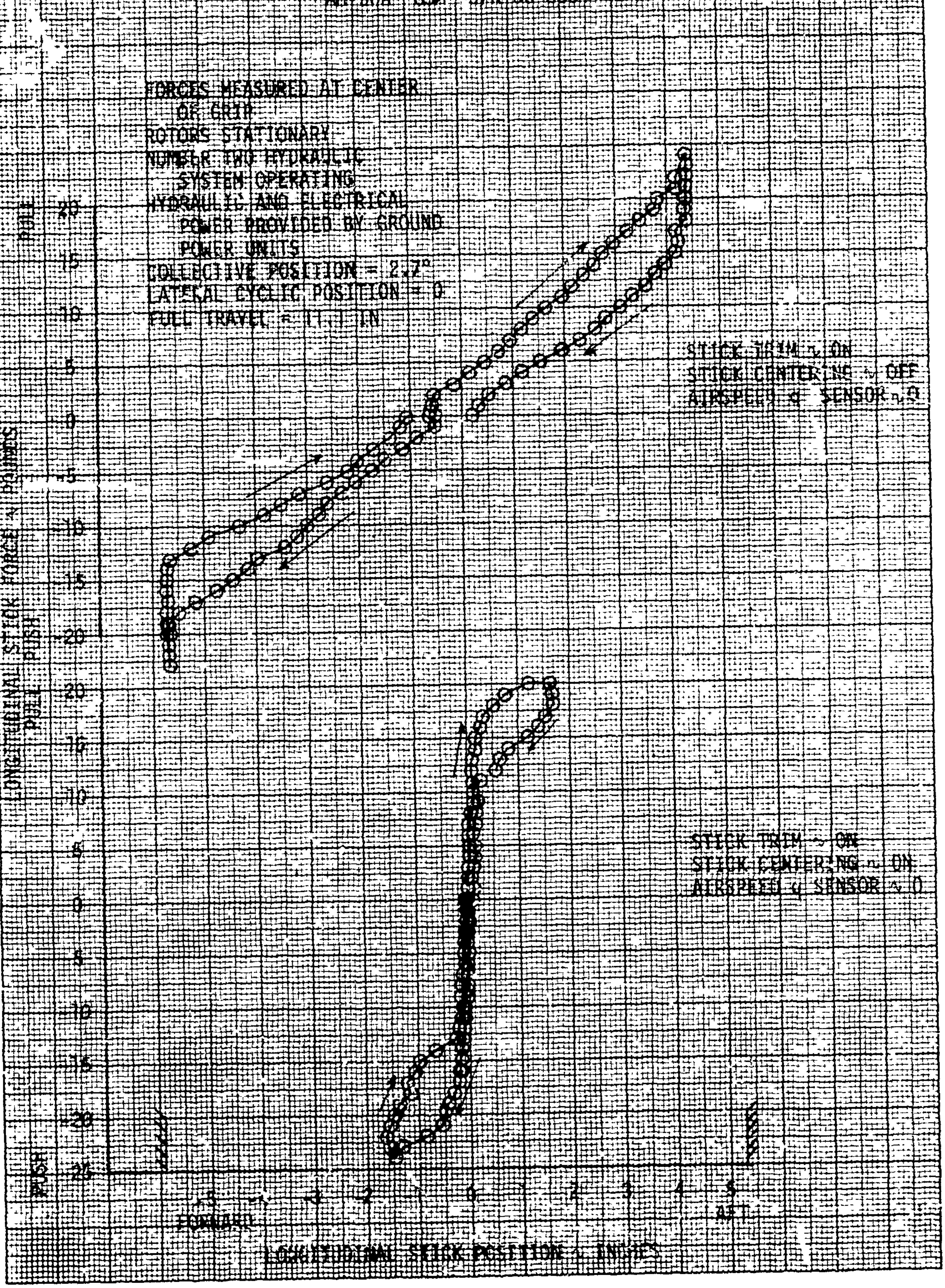
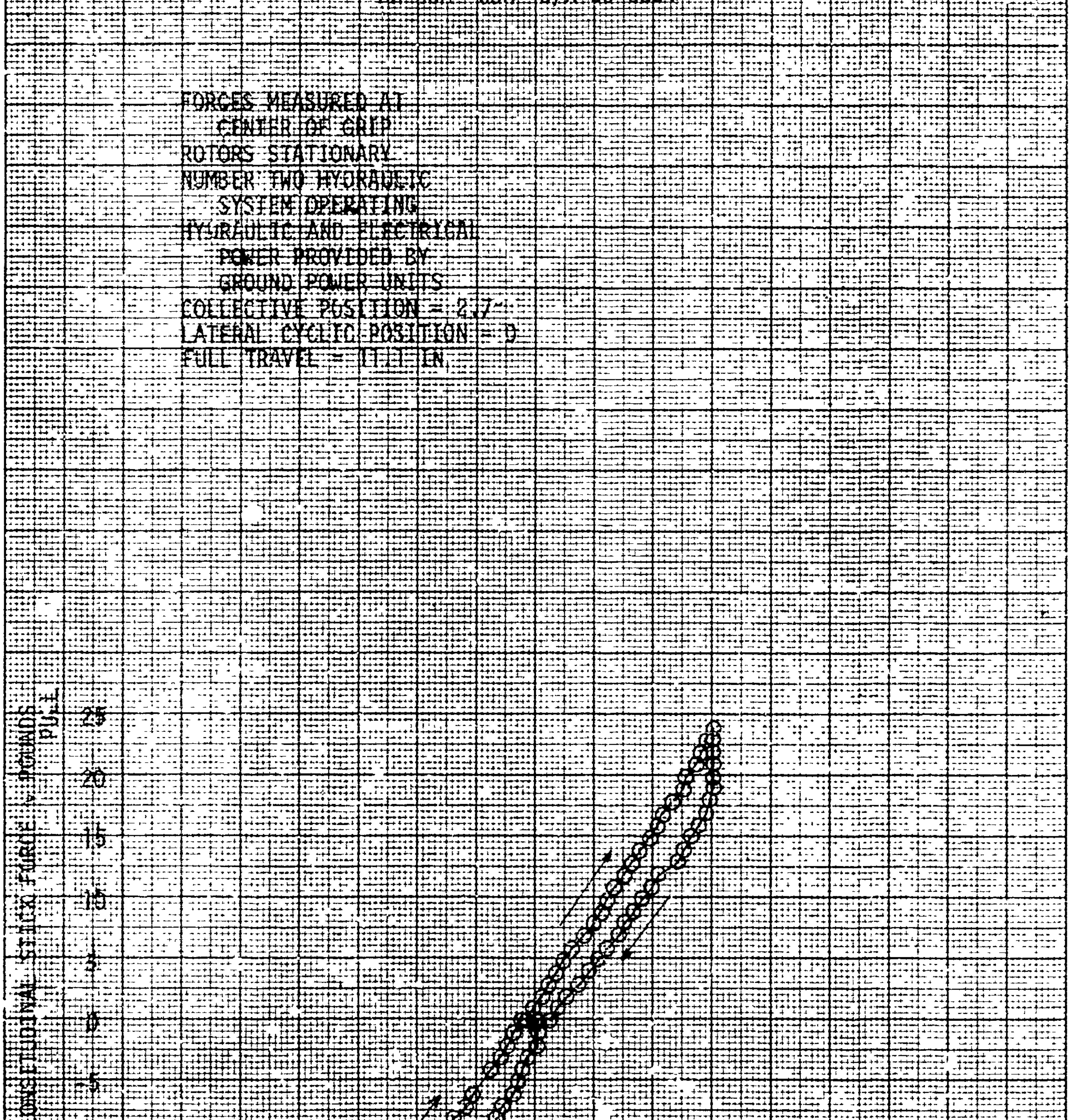


FIGURE 96
LONGITUDINAL STICK FORCES
AH-56A USA S/N 66-8834



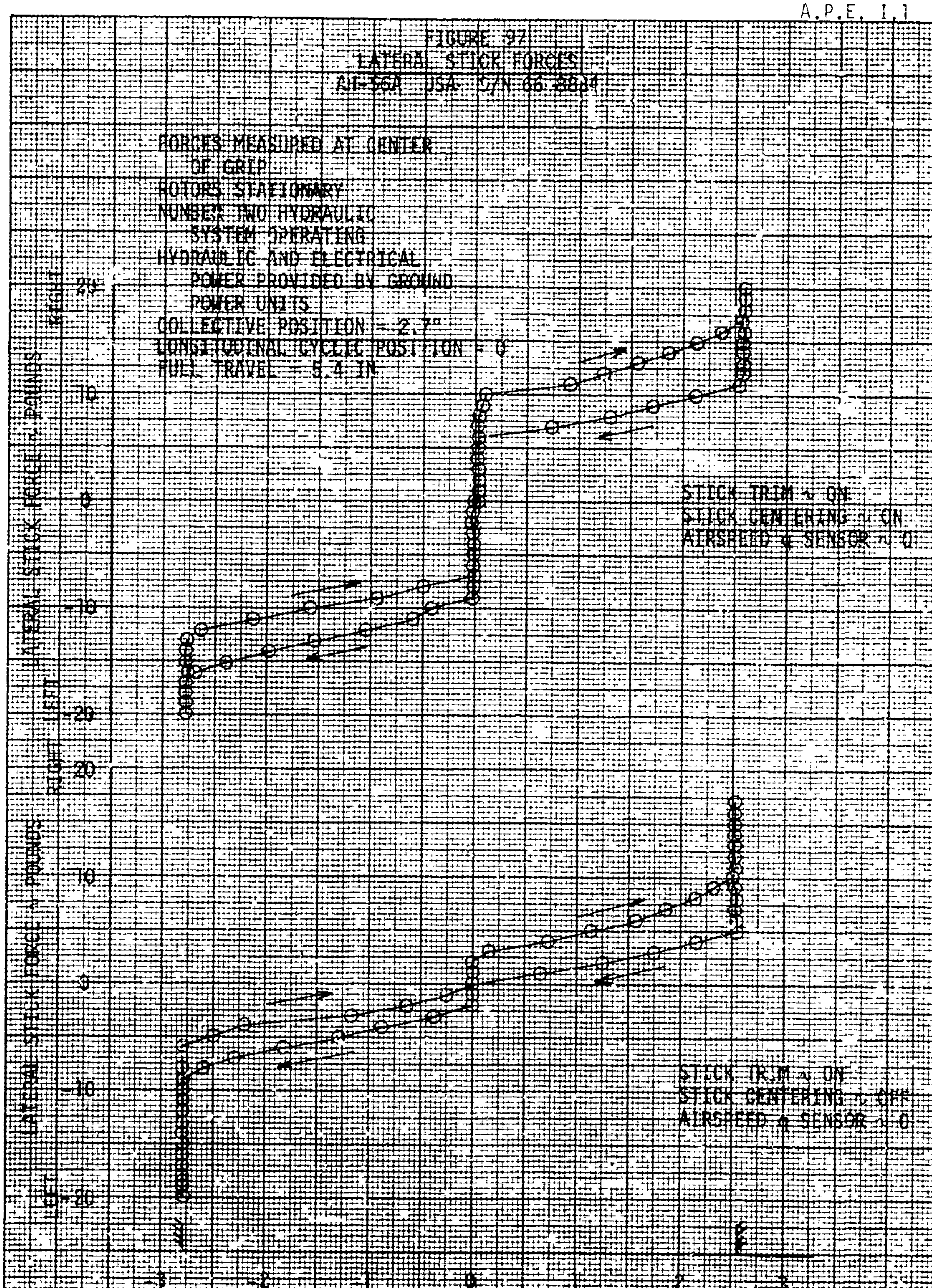


FIGURE 97
LATERAL STICK FORCES
AH-56A USA S/N 66-8834
FORCES MEASURED AT CENTER OF GRIP
ROTORS STATIONARY
NUMBER TWO HYDRAULIC SYSTEM OPERATING
HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS
COLLECTIVE POSITION = 2.7°
LONGITUDINAL CYCLIC POSITION = 0
FULL TRAVEL = 5.4 IN
STICK TRIM ~ ON
STICK CENTERING ~ ON
AIRSPEED SENSOR ~ 0
STICK TRIM ~ ON
STICK CENTERING ~ OFF
AIRSPEED SENSOR ~ 0
RIGHT
LEFT
LATERAL STICK FORCE ~ POUNDS
20
10
0
-10
-20
-3
LEFT
-2
-1
0
1
2
3
RIGHT
LATERAL STICK POSITION ~ INCHES

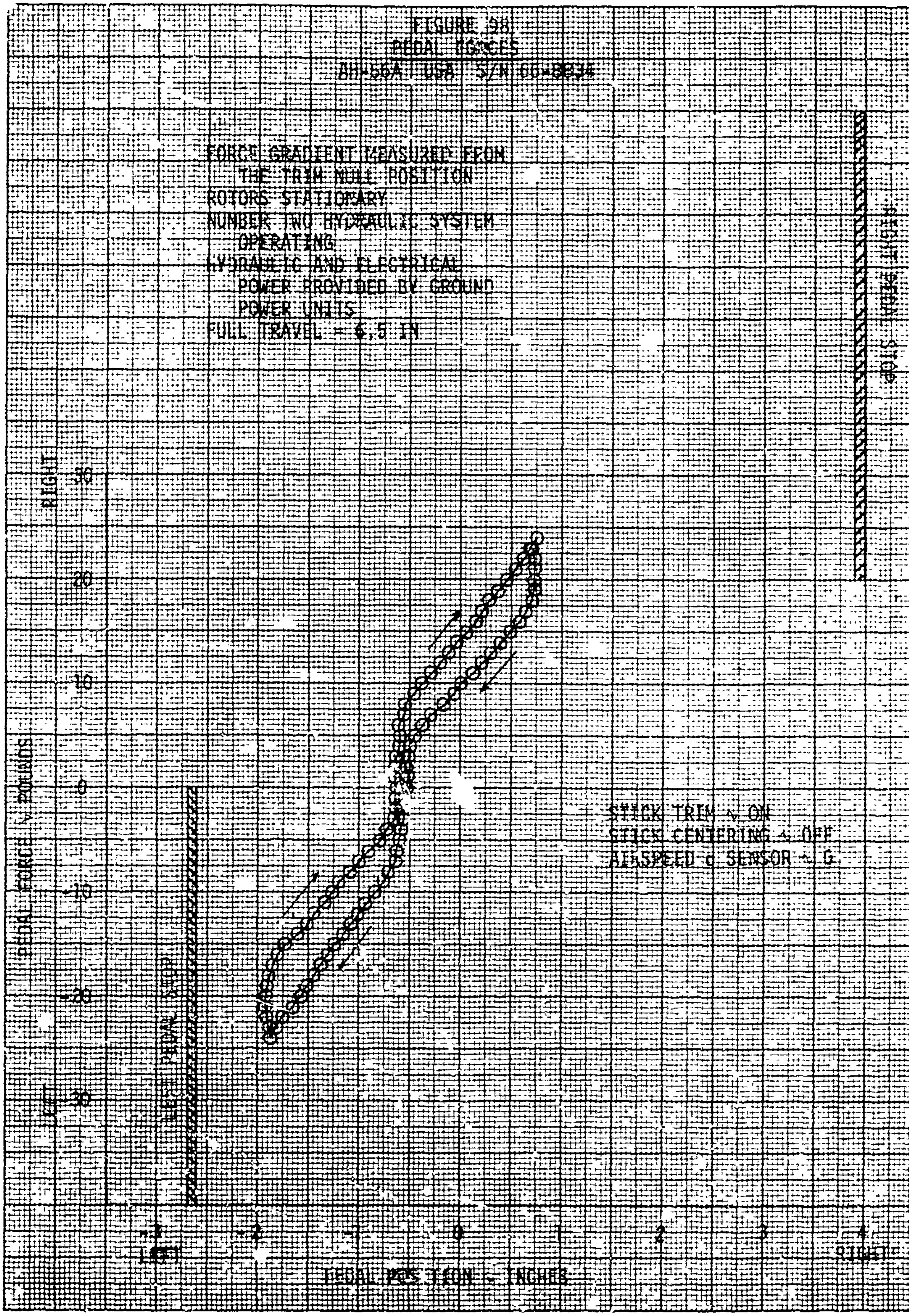
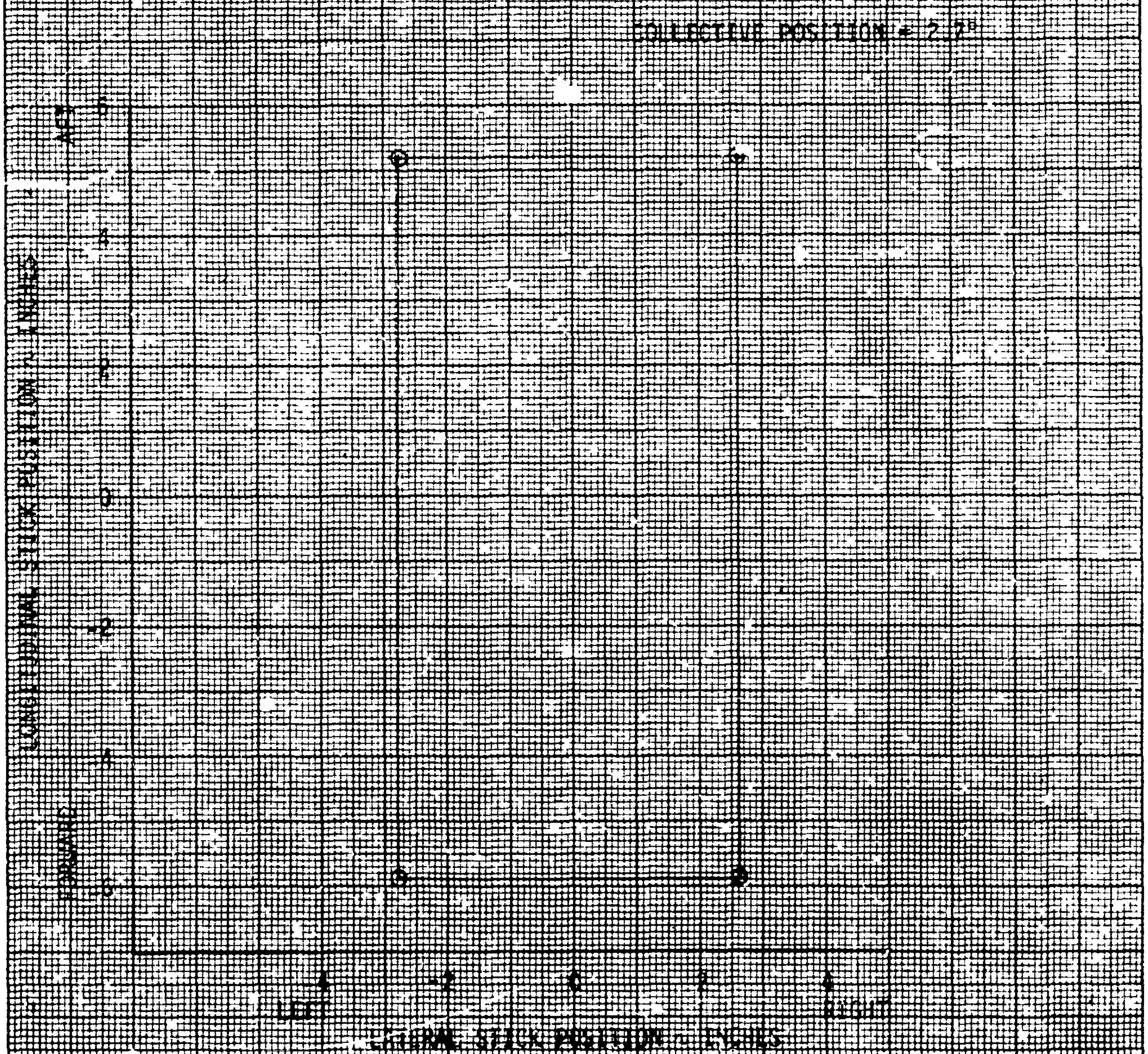


FIGURE 98
PEDAL FORCES
AH-56A USA S/N 66-8834
FORCE GRADIENT MEASURED FROM THE TRIM NULL POSITION
ROTORS STATIONARY
NUMBER TWO HYDRAULIC SYSTEM OPERATING
HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS
FULL TRAVEL = 6.5 IN
RIGHT PEDAL STOP
LEFT PEDAL STOP
STICK TRIM ~ ON
STICK CENTERING ~ OFF
AIRSPEED δ SENSOR ~ G
PEDAL FORCE ~ POUNDS
RIGHT
LEFT
30
20
10
0
-10
-20
-30
-3
-2
-1
0
1
2
3
4
LEFT
RIGHT
PEDAL POSITION ~ INCHES

FIGURE 99
LIMITS OF CYCLIC STICK TRAVEL
AH-56A USA S/N 66-8828



APPENDIX G. APE I.2 TEST DATA

This appendix includes the test data obtained during APE I.2. No rotor or control system changes were made from the APE I.1 configurations. The tests were conducted at the following nominal conditons: 4000-foot density altitude, 100-percent rotor speed, 18,300-pound gross weight, station 300 cg location (defined as mid during APE I.2) and in the external stores configuration – two ballasted XM159 rocket pods on the outboard stations and two empty TOW pods on the inboard stations. One significant limit applied to APE I.2 testing was a maximum collective angle of 12 degrees, which precluded hovering.

INDEX

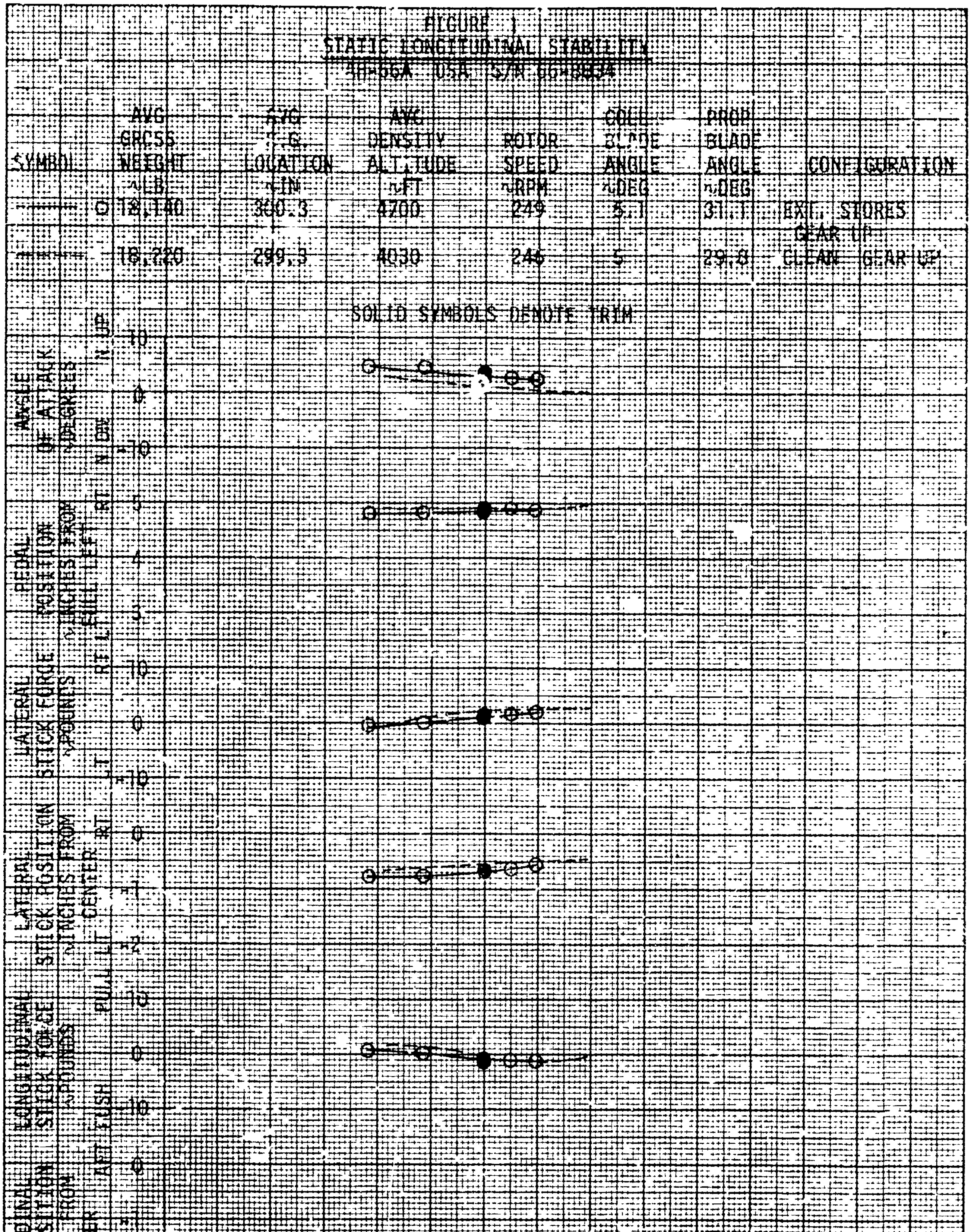


FIGURE 1
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834
SYMBOL
AVG GROSS WEIGHT ~LB
AVG C.G. LOCATION ~IN
AVG DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL BLADE ANGLE ~DEG
PROP BLADE ANGLE ~DEG
CONFIGURATION
18,140 300.3 4700 249 5.1 31.1 EXT. STORES GEAR UP
18,220 299.3 4030 246 5 29.8 CLEAN GEAR UP
SOLID SYMBOLS DENOTE TRIM
ANGLE OF ATTACK ~DEGREES
N UP
N DN
PEDAL POSITION ~INCHES FROM FULL LEFT
RT L
LATERAL STICK FORCE ~POUNDS
RT
LATERAL STICK POSITION ~INCHES FROM CENTER
CENTER RT
FULL LT
LONGITUDINAL STICK FORCE ~POUNDS
PULL
AFT PUSH
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER
CENTER
FWD
10
0
-10
5
4
3
10
0
-10
0
-1
-2
10
0
-10
0
-1
-2
-3
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 2
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	AVG COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KTS	CONFIGURATION
———O	17,860	300.1	3840	247	9.3	11.9	61	EXT. STORES GEAR DOWN
—·—·—	17,850	299	3900	245	9.4	14.4	59	CLEAN GEAR DOWN

SOLID SYMBOLS DENOTE TRIM
APE 1.1 CONTROL FORCE DATA NOT AVAILABLE.

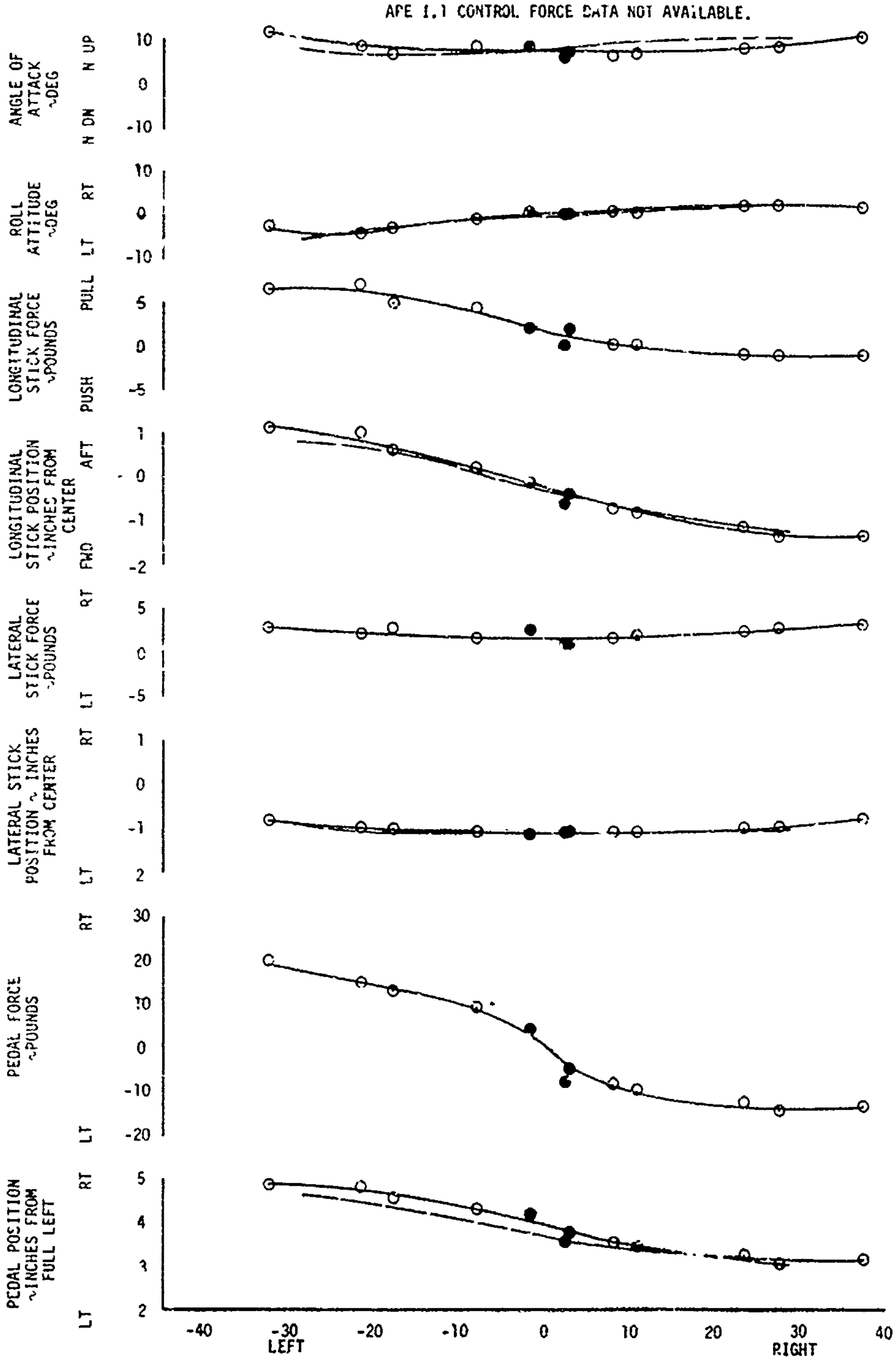


FIGURE 3
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	PROP BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KTS	CONFIGURATION
——O	17,970	300.7	4360	248	7.5	18.7	103	EXT. STORES GEAR UP
— — —	18,450	299.6	4540	248	7	18.5	99	CLEAN GEAR UP

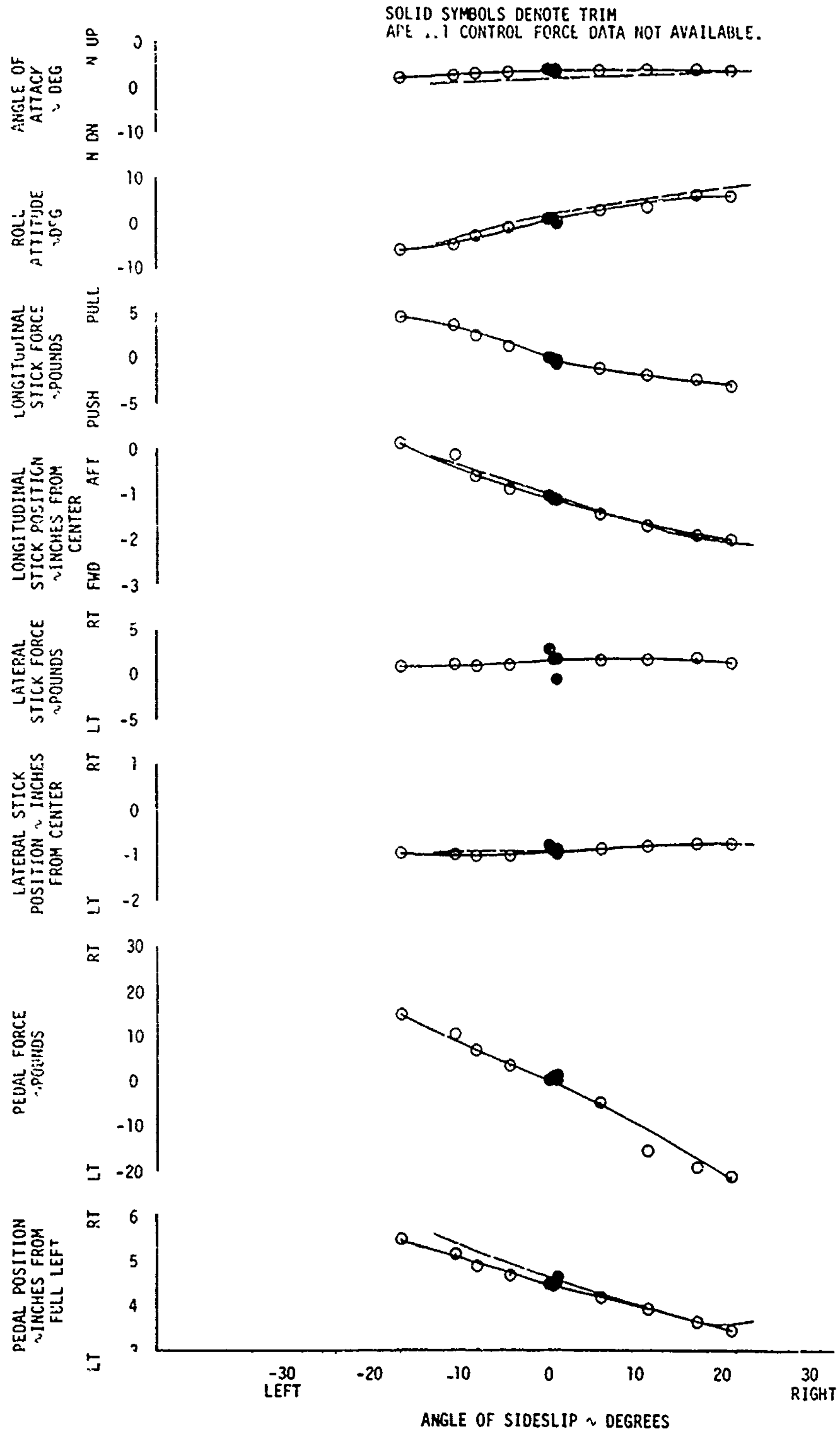
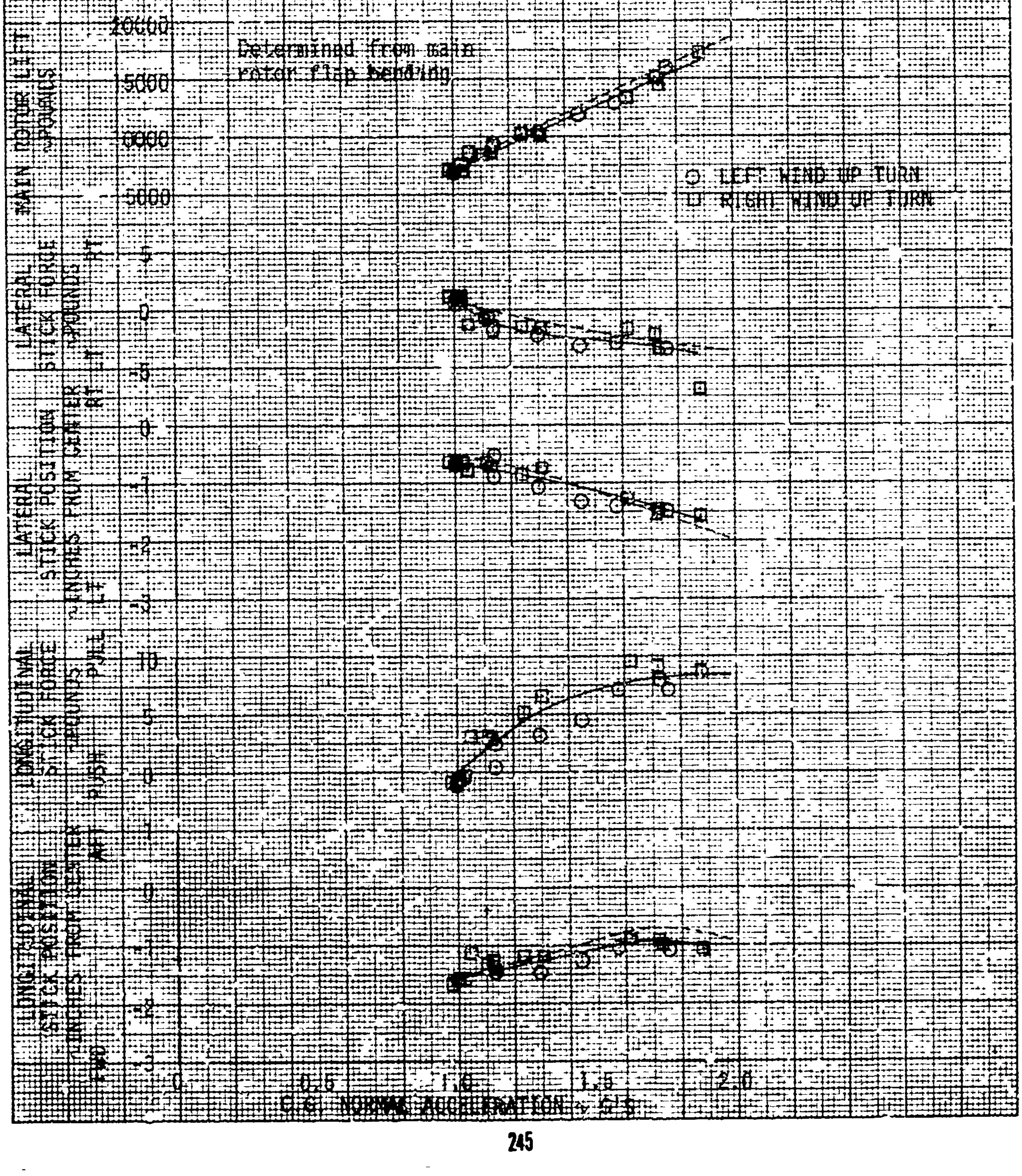


FIGURE 4
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
○	18,110	300.3(MID)	3770	248	5.2	150	EXT STORES GEAR UP
□	17,930	300.3(MID)	3540	248	5.2	150	EXT STORES GEAR UP
	16,200	299.4(MID)	3880	247	5.0	154	CLEAN GEAR UP
	16,648	299.8(MID)	4200	246	5.1	156	CLEAN GEAR UP



APPENDIX H. APE I.3A TEST DATA

This appendix contains test data obtained during APE I.3A. Prior to APE I.3A, the main rotor droop was increased to 3 degrees and 10 minutes. A pitch desensitizer and pitch/roll decoupler were added to the longitudinal control system. Also, the maneuver gradient design effectiveness was changed from 6 to 9 lb/g. The roll desensitizer was removed, and the roll compensator was added to the lateral control system. Additionally, the right wing and horizontal stabilizer incidence were changed, and detuning weights were removed from the left horizontal stabilizer. All stability and control augmentation devices except for the lift/roll decoupler were activated for these tests. Tests were conducted at the following nominal conditions: 5000-foot density altitude, 100-percent rotor speed, 18,300- to 20,500-pound gross weight, station 300 cg location (defined as aft for APE I.3), and in both the clean and external stores configuration. The gross weight, center-of-gravity location, and density altitude stated on the static longitudinal stability, static lateral-directional stability, maneuvering stability, and controllability figures are average values for the data presented.

INDEX

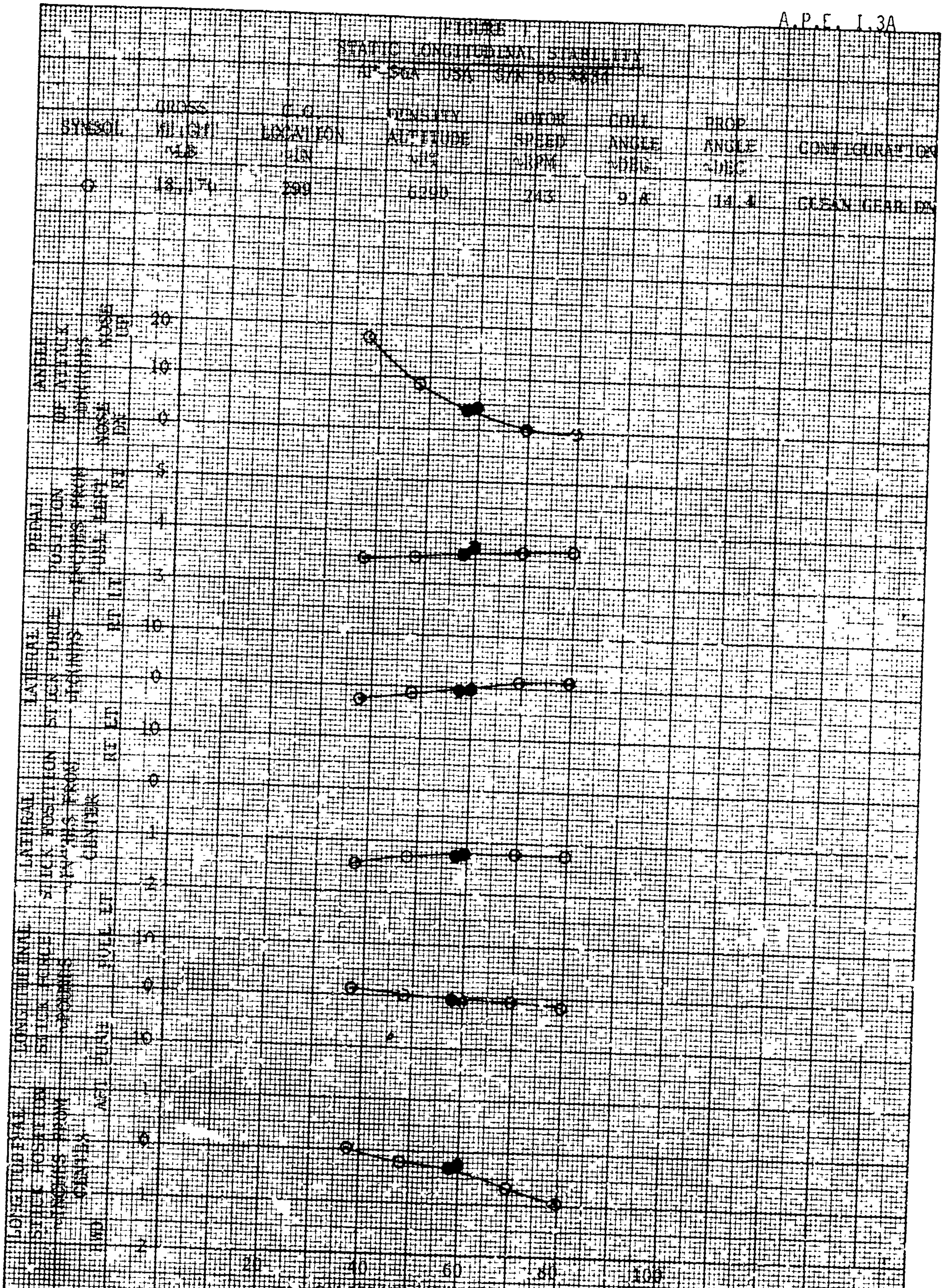


FIGURE 1
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8830
SYMBOL
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL ANGLE ~DEG
PROP ANGLE ~DEG
CONFIGURATION
18,174
299
6290
243
9.8
14.4
CLEAN GEAR DN
ANGLE OF ATTACK ~DEGREES
NOSE UP
NOSE DN
PEDAL POSITION ~INCHES FROM FULL LEFT
LATERAL STICK FORCE ~POUNDS
RT LT
LATERAL STICK POSITION ~INCHES FROM CENTER
FULL LT
LONGITUDINAL STICK FORCE ~POUNDS
PULL
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER
FWD AFT
CALIBRATED AIRSPEED ~ KNOTS
20
40
60
80
100

FIGURE 2
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
○	20,180	299.2	5750	247	9.9	13.7	EXT. STORES GEAR DN

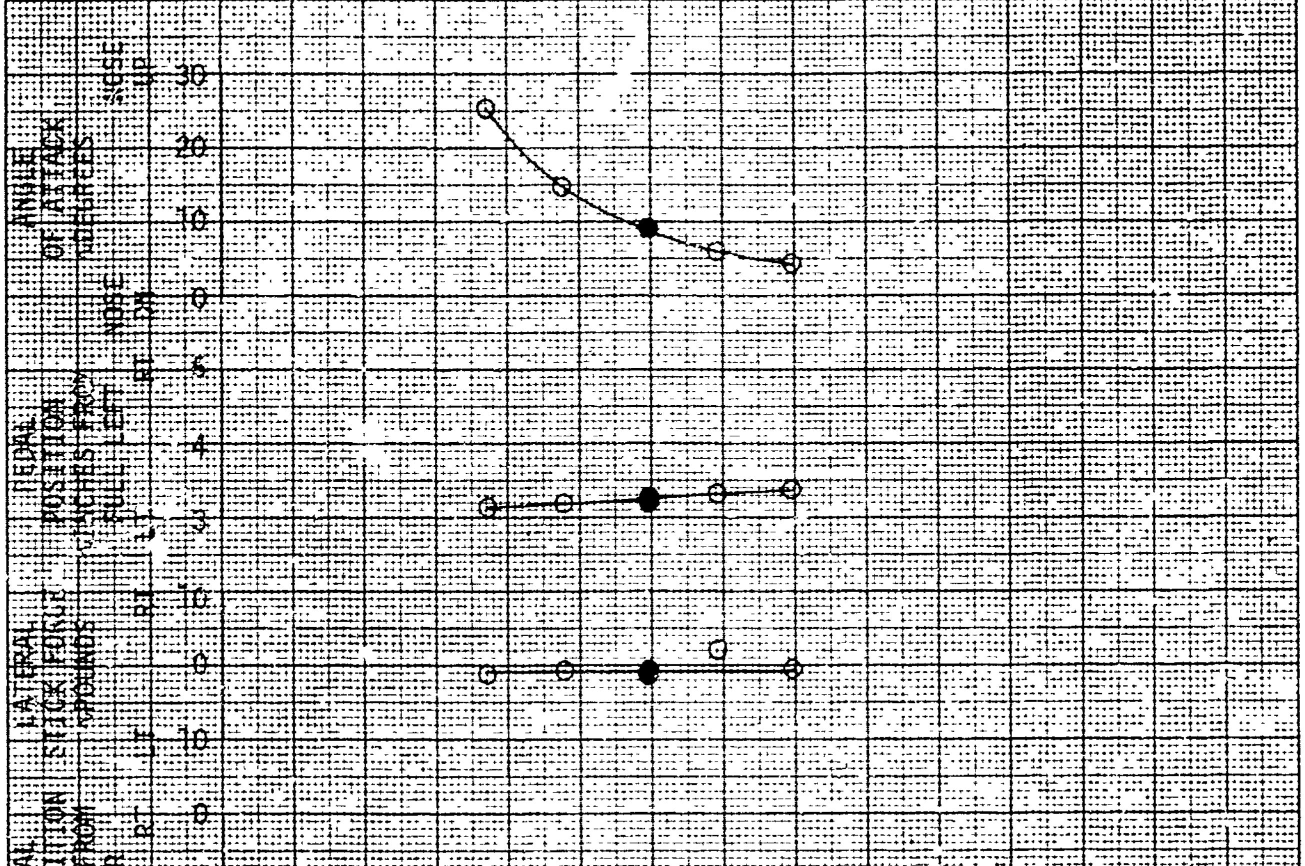


FIGURE 5
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
○	17,980	298.9	5440	245	7.2	18.6	CLEAN GEAR DN

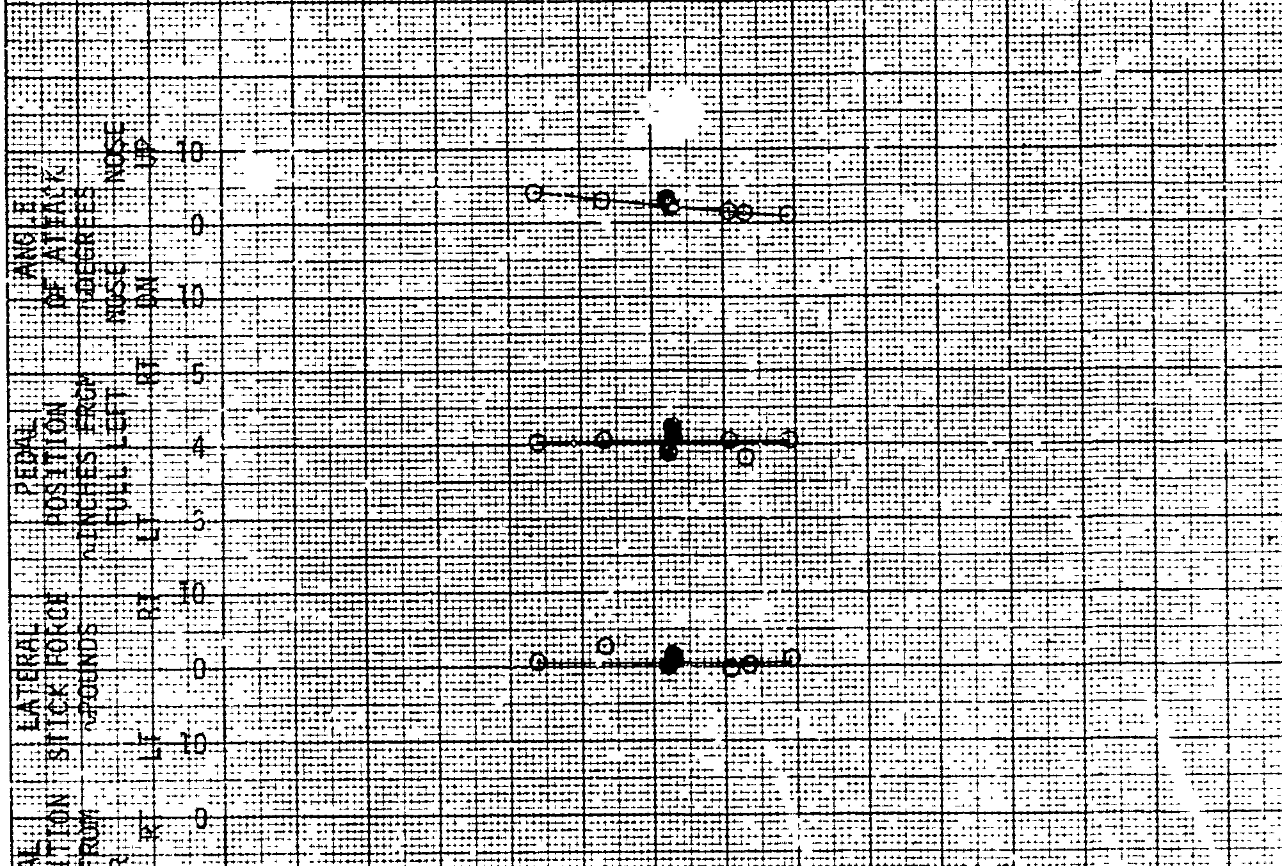


FIGURE 4
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
○	17,800	299.8	5180	246	3.7	29.5	CLEAN GEAR UP

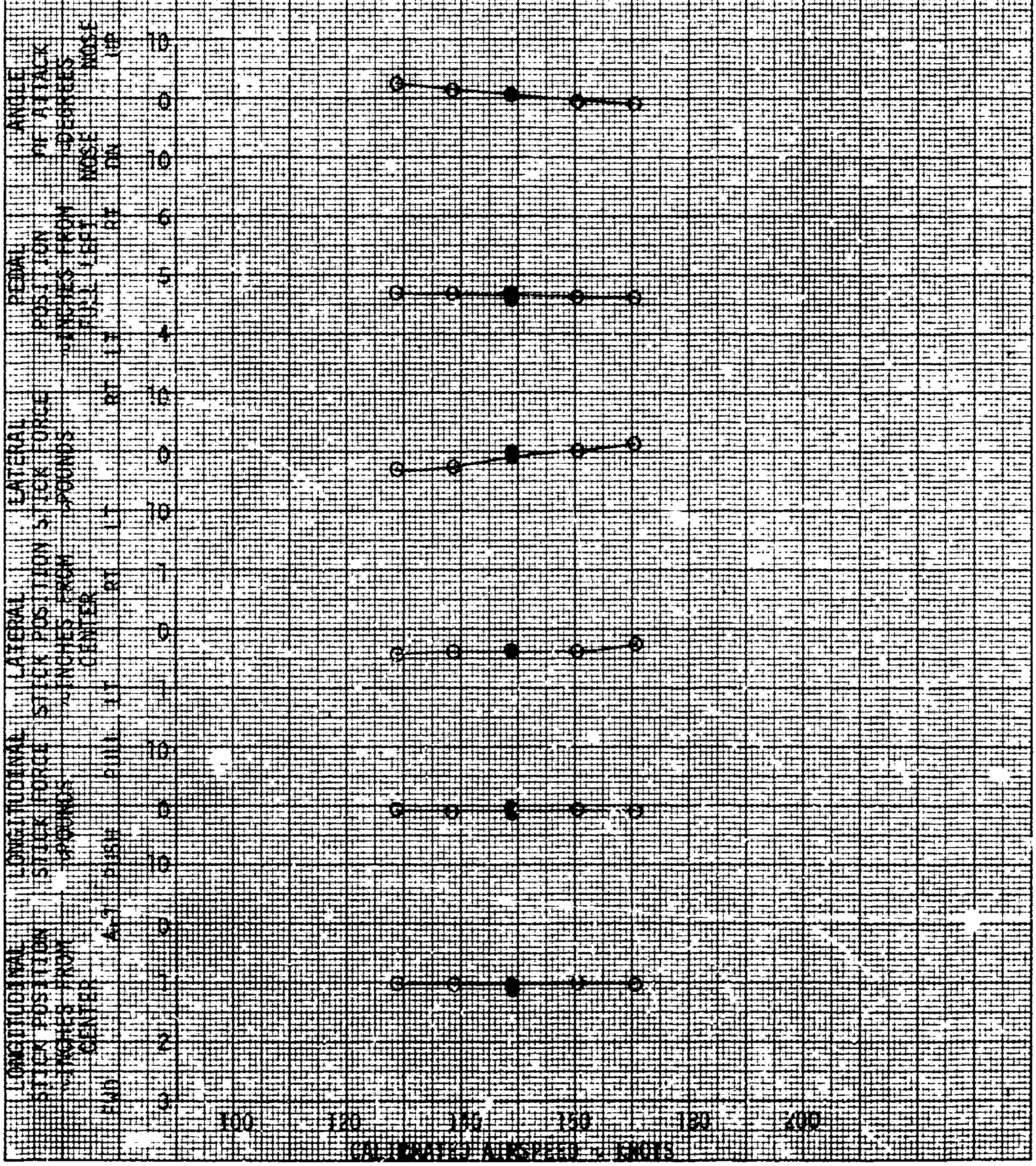


FIGURE 5

STATIC LONGITUDINAL STABILITY

AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
○	18,190	299.9	5440	247	5.8	28.5	CLEAN GEAR UP

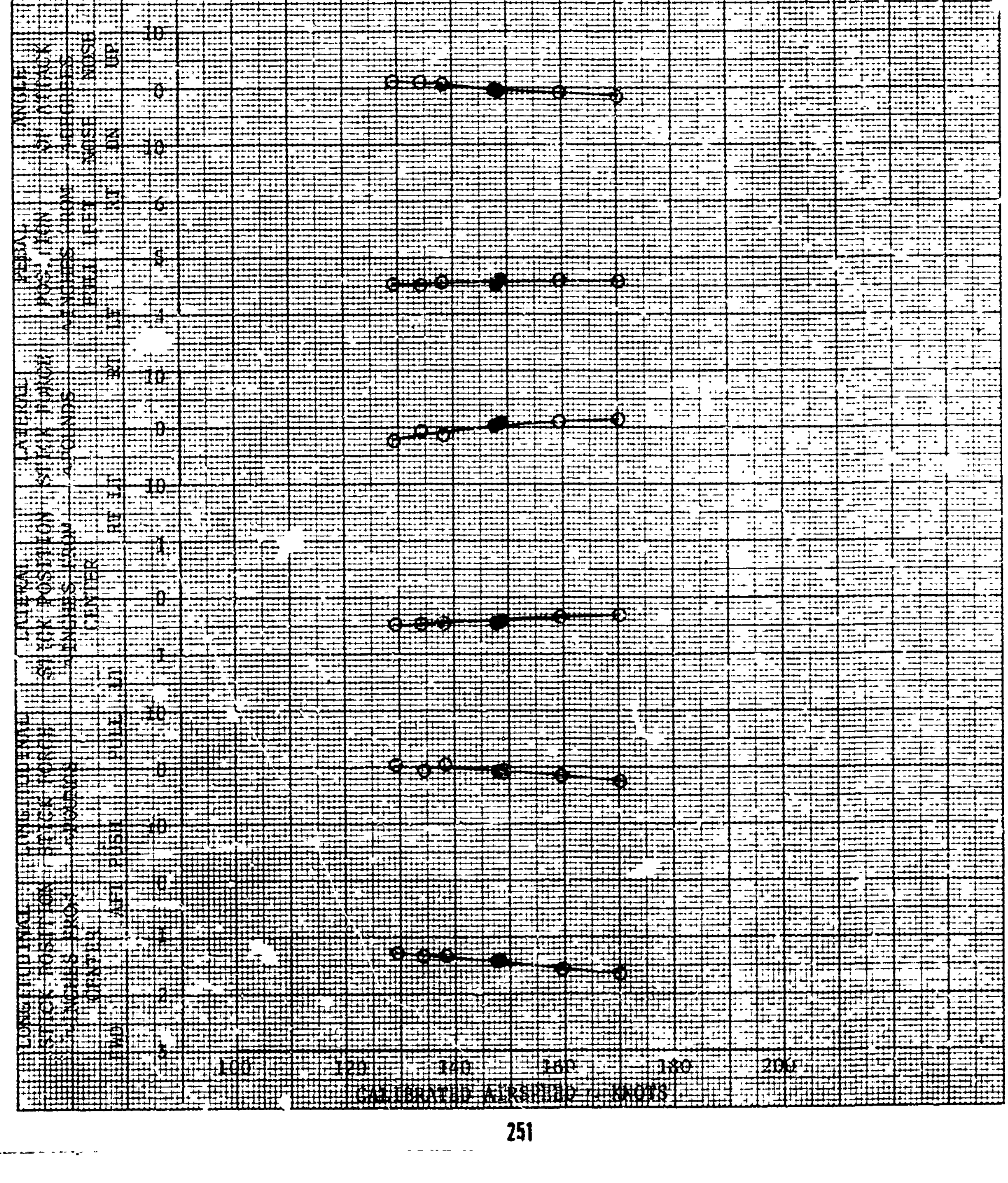


FIGURE 6
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
O	20,060	300	4830	256	5.7	34.9	EXT STORES GEAR UP

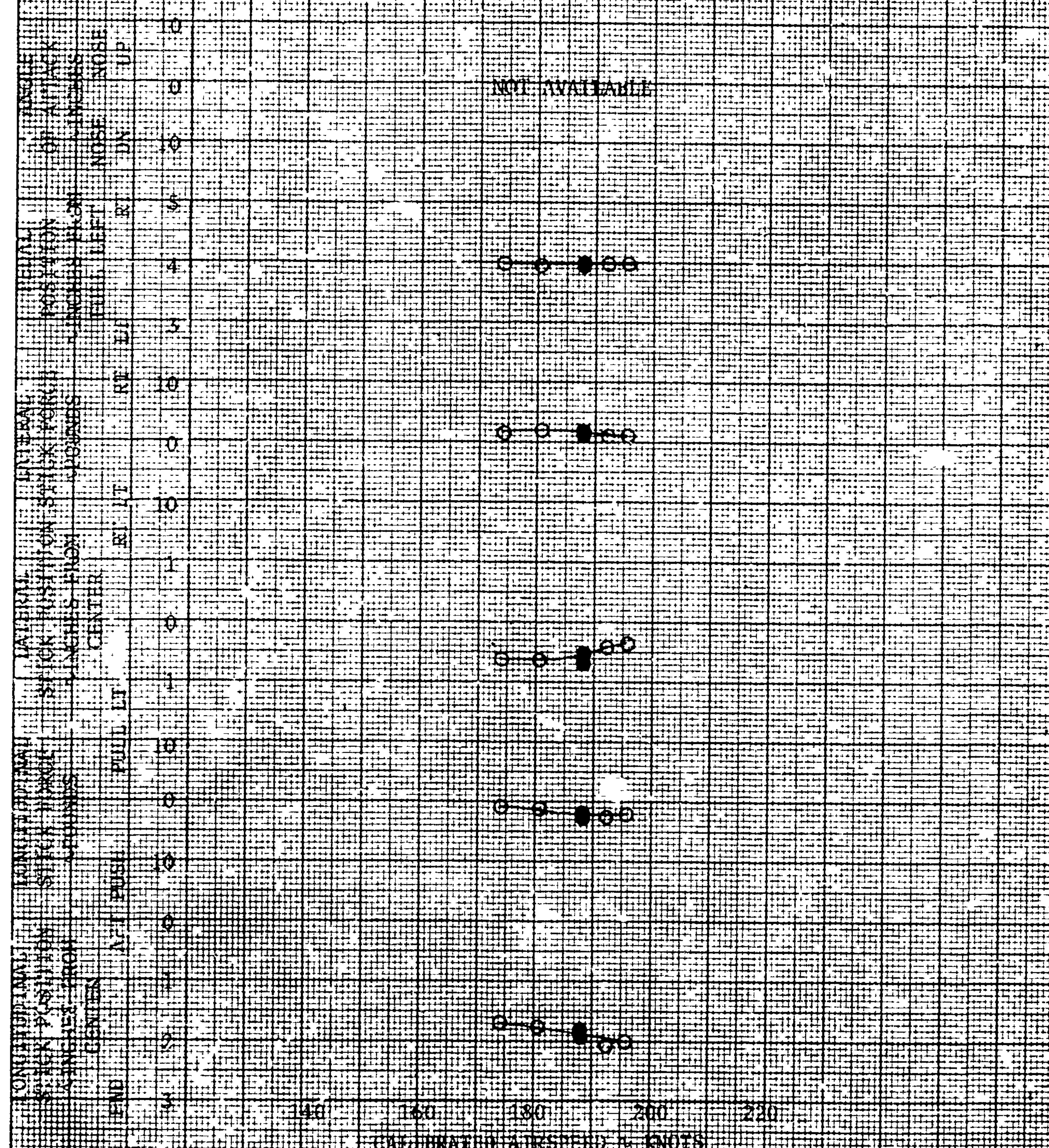


FIGURE 7
RELEASE OF LONGITUDINAL CONTROL TO TRIM
AH-56A USA S/N 66-8834

LONGITUDINAL, PITCH & ALTITUDE ———
LATERAL, ROLL & AIRSPEED - - - - - -
PEDAL, SIDESLIP & TEMPERATURE — - —

TRIM AIRSPEED ~KCAS	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION
101	18020	298.9	5800	277	CLEAN GEAR UP

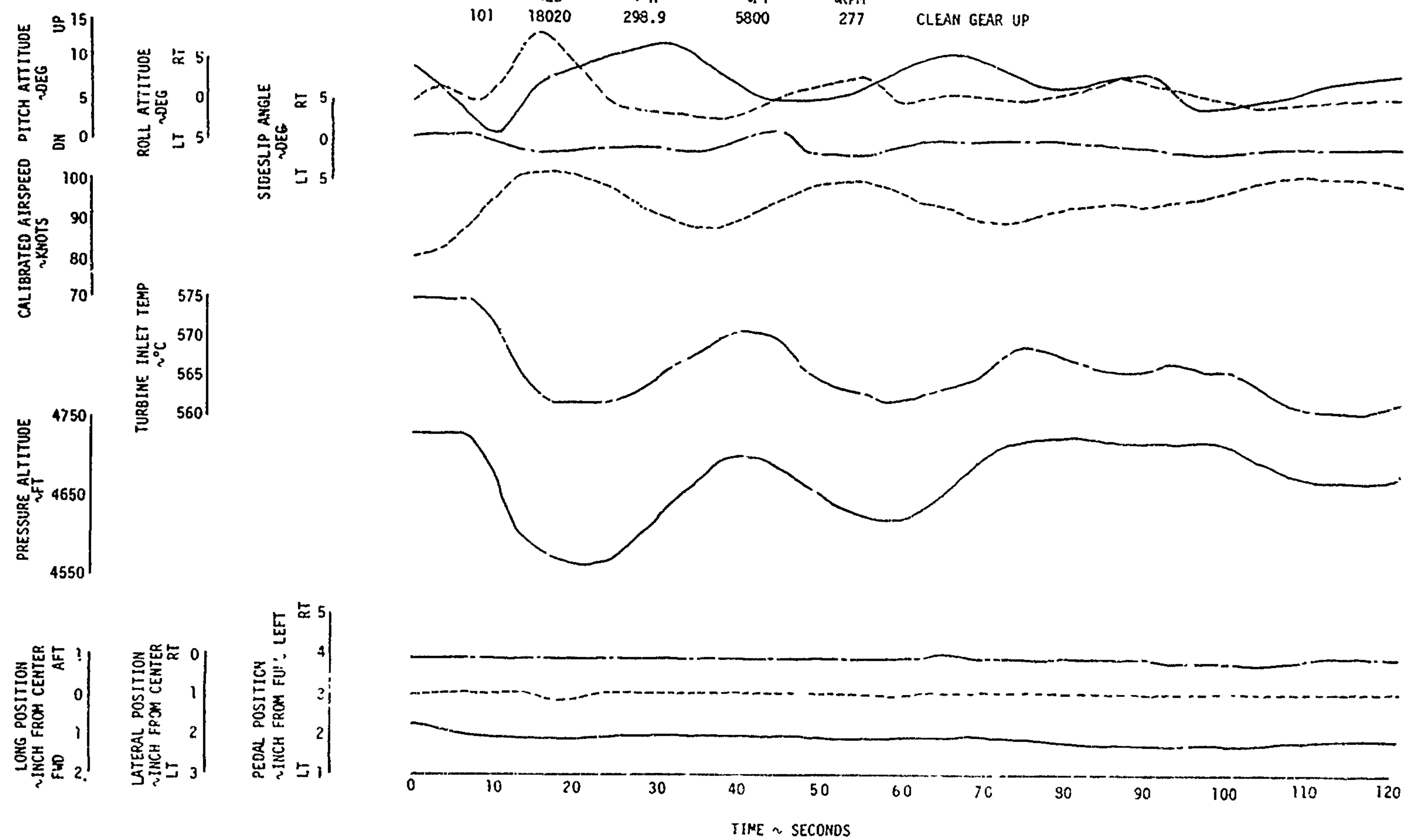


FIGURE 8
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	ROLL ANGLE ~DEG	PROP ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION
○	17,980	298.9	6530	244	9.9	14.2	59	CLEAN GEAR DN

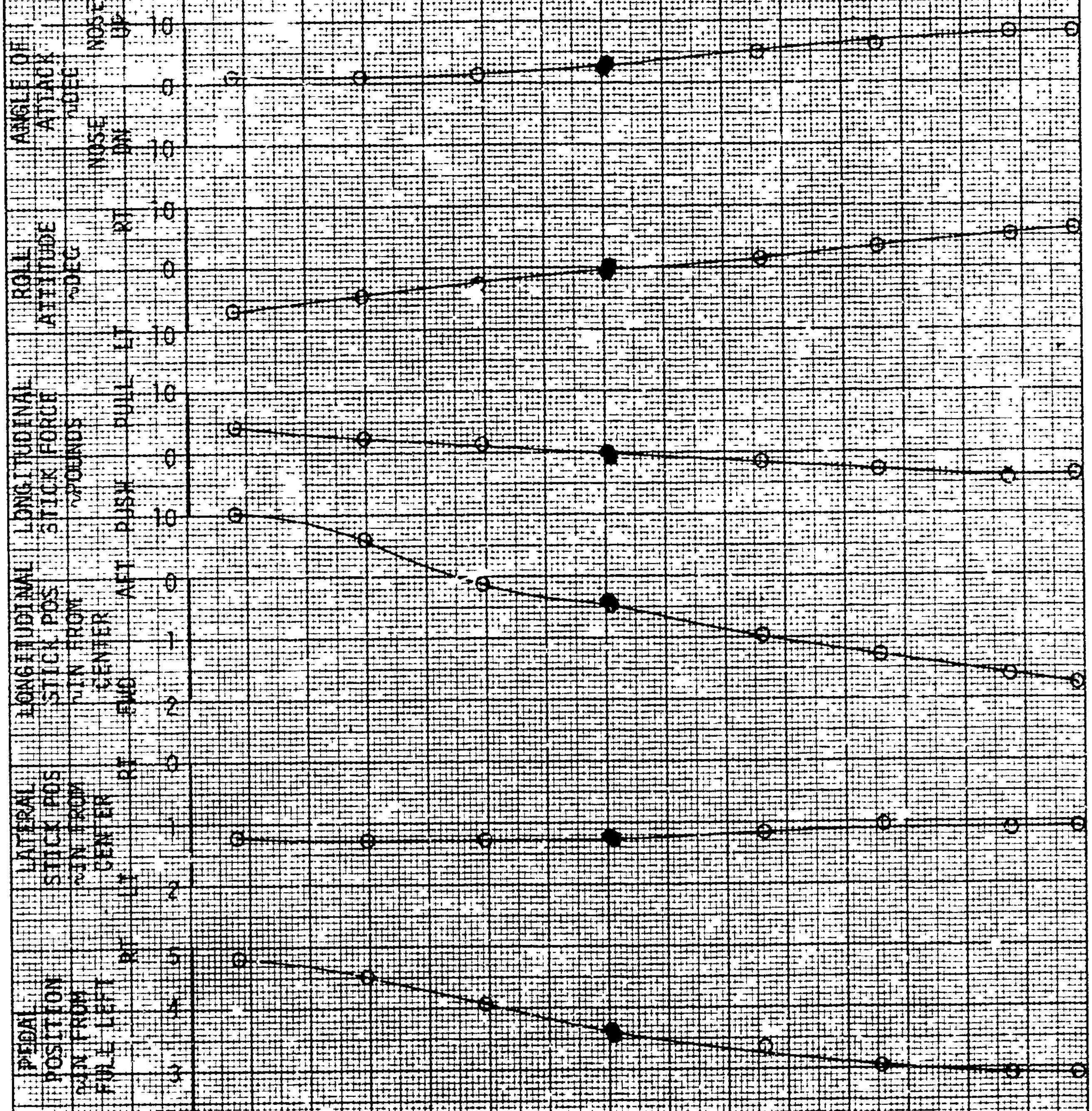


FIGURE 9
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION
○	18,330	299.9	5710	245	7.8	18.6	01	CLEAN GEAR UP

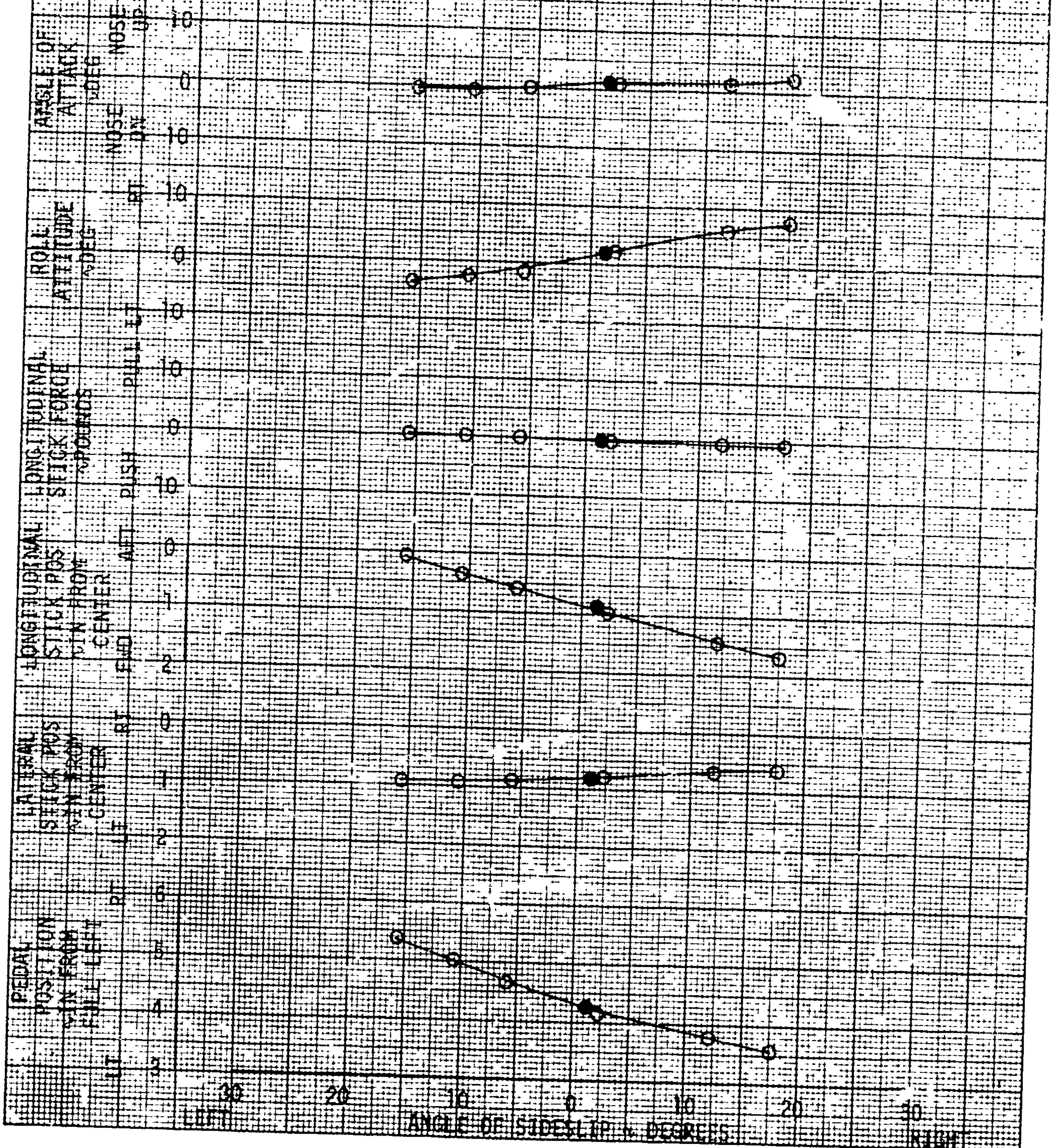


FIGURE 10
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION
○	18,050	299.8	5130	246	5.9	28.6	149	CLEAN GEAR UP

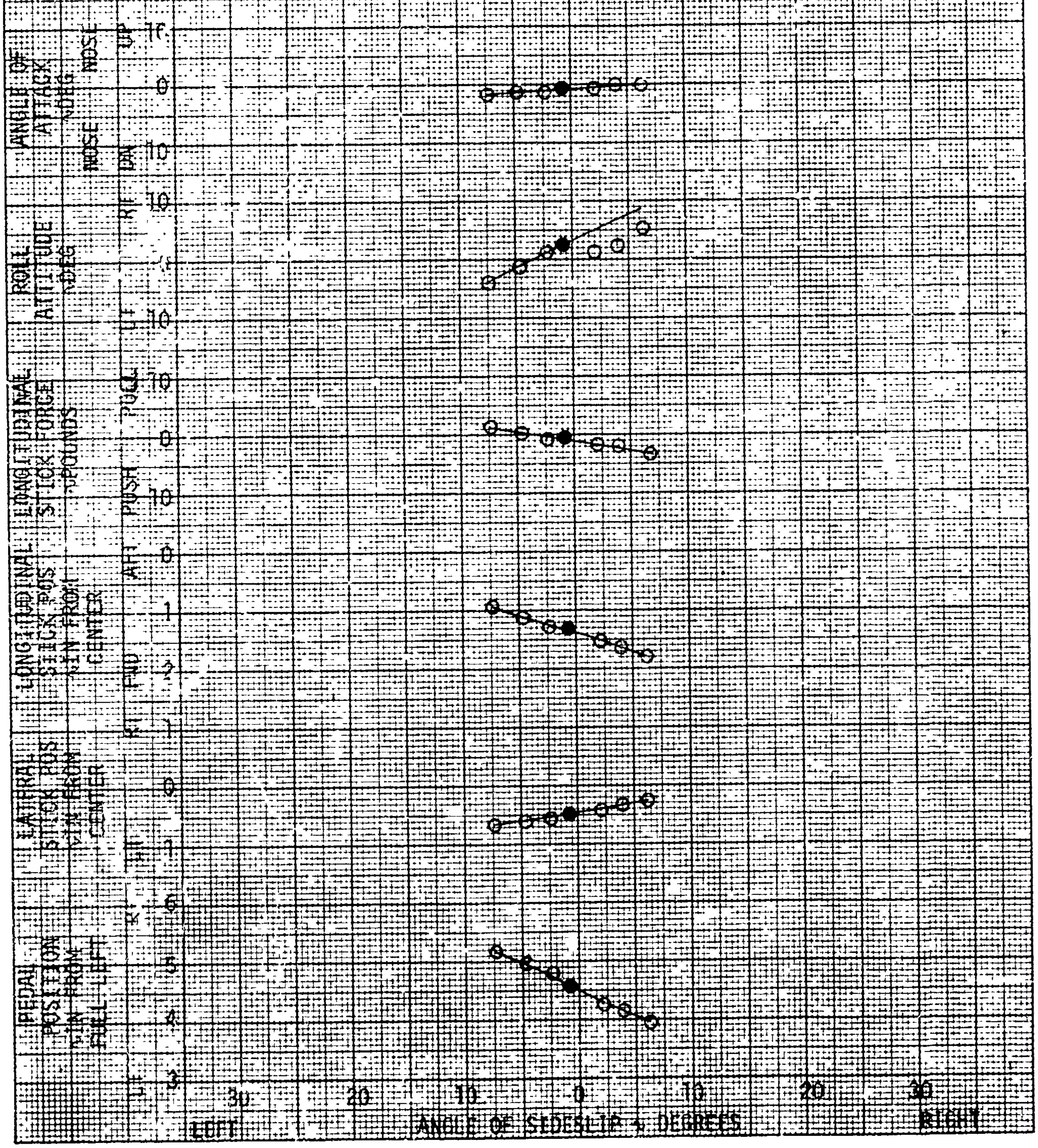
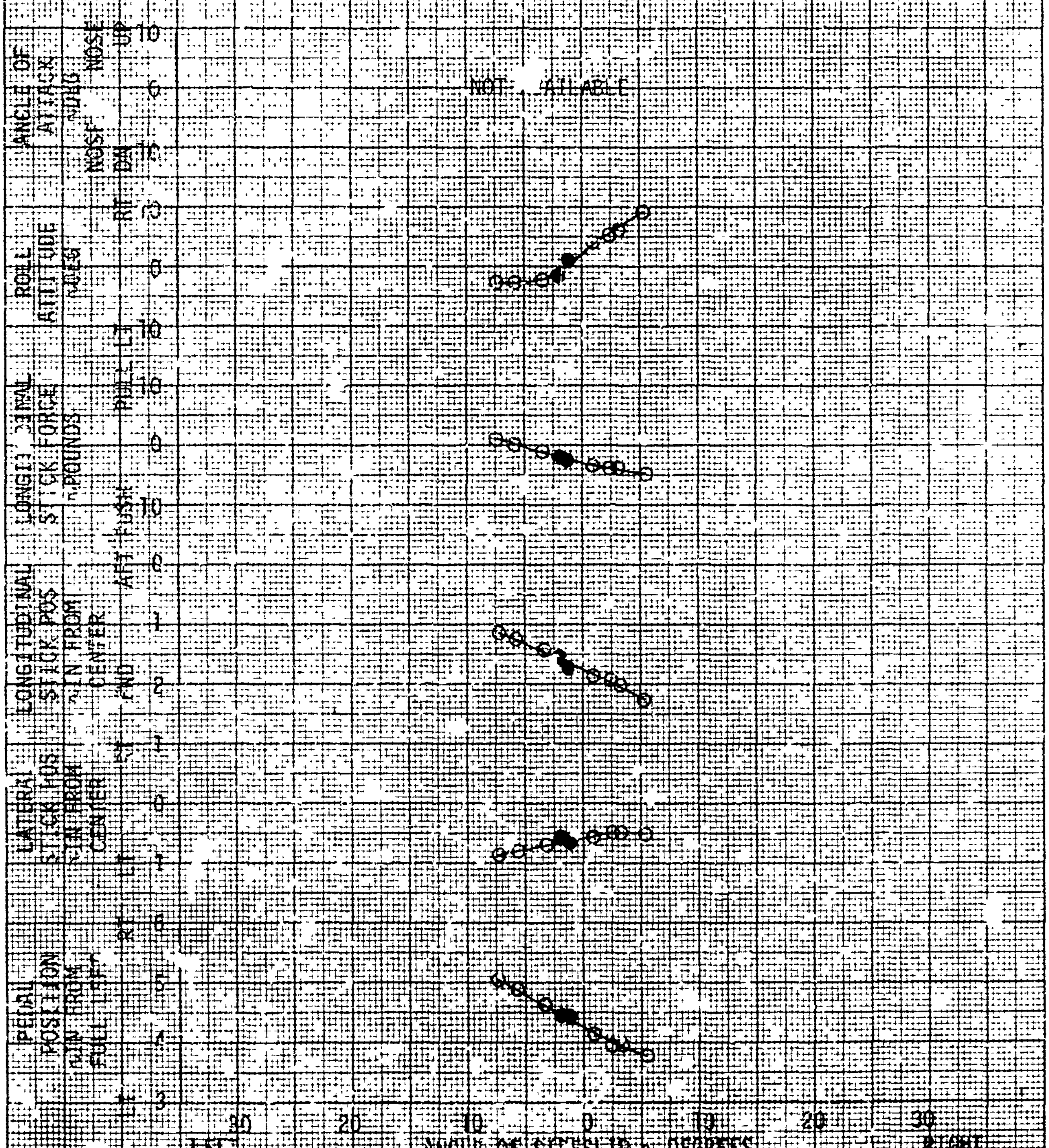


FIGURE 11
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION
○	19,740	299.9	544[illegible]	[illegible]	5.3	38.6	172	EXT STORES GEAR UP



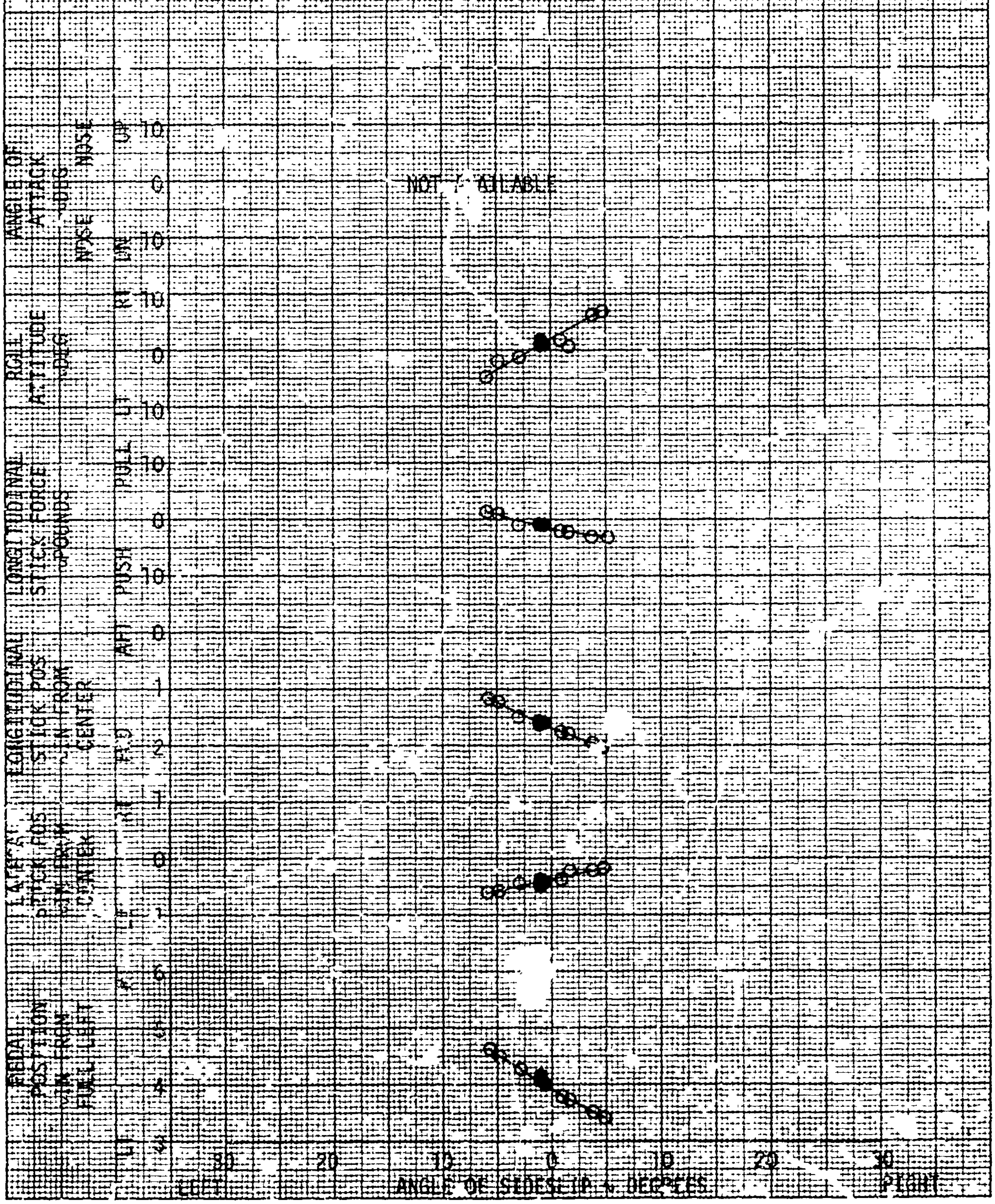
A.P.F. 1.3A

FIGURE 12

STATIC LATERAL-DIRECTIONAL STABILITY

AH-56A USA S/N 66-8830

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION
○	18,460	299.9	5350	247	6.9	33	180	CLEAN GEAR UP



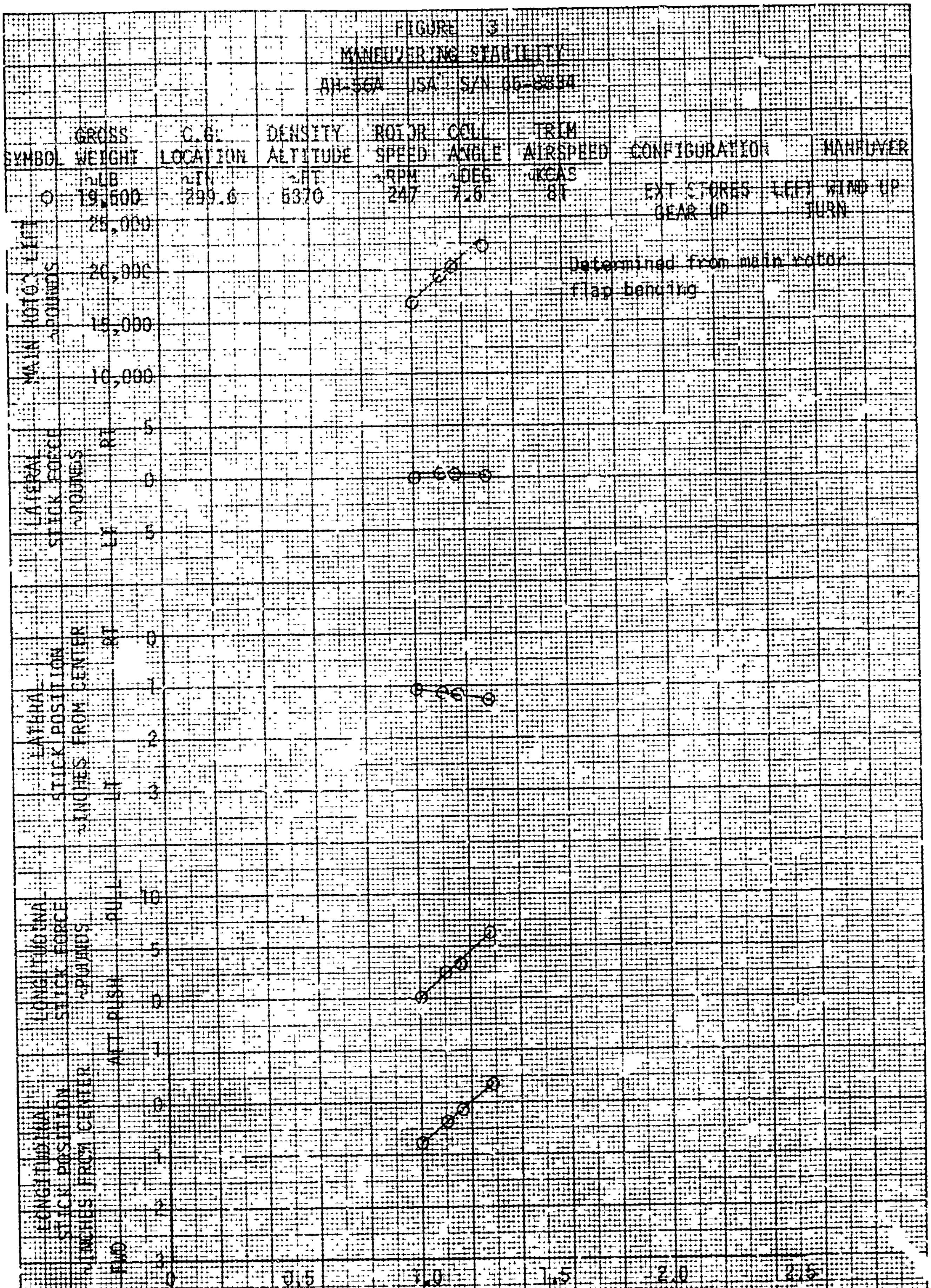
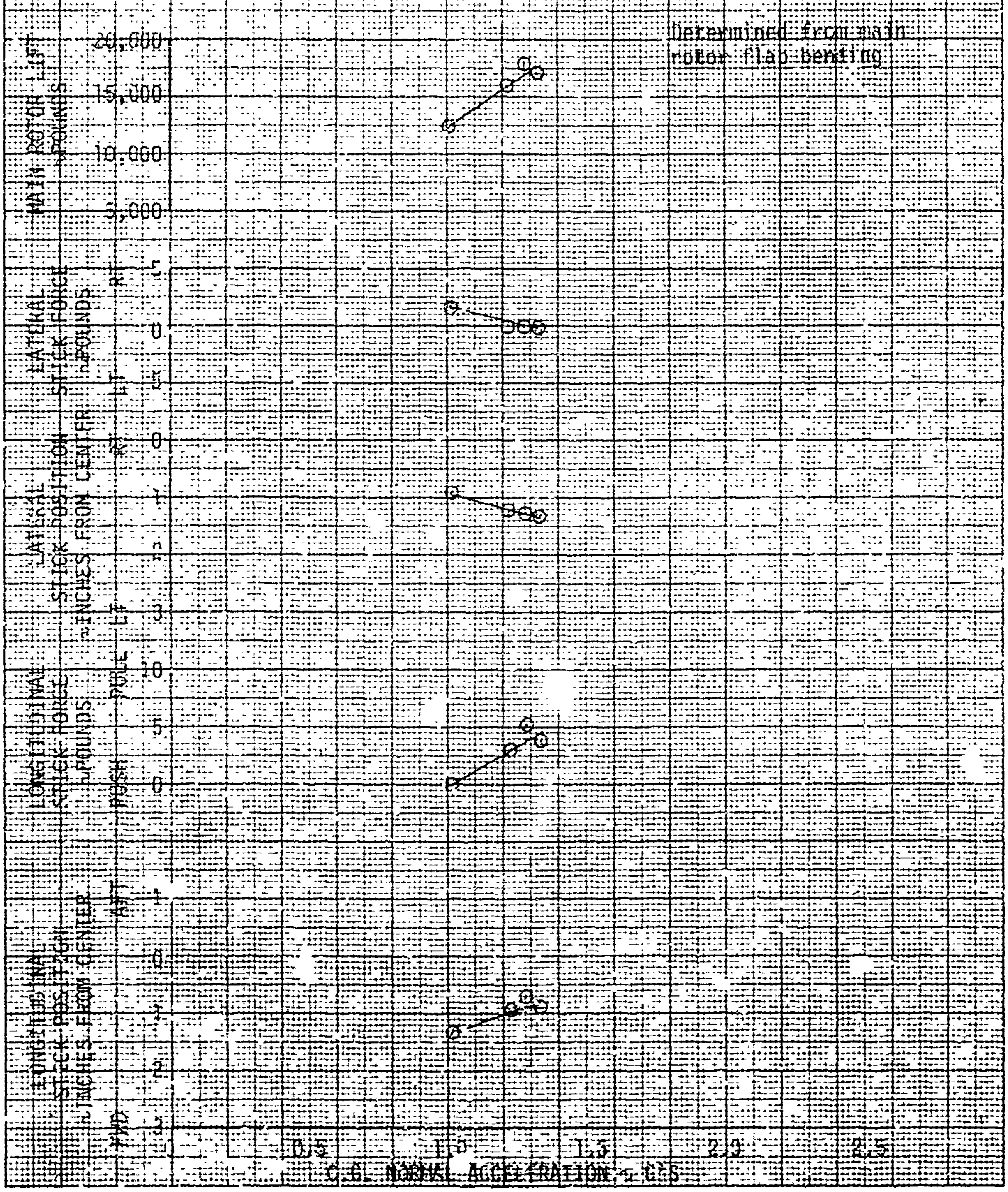


FIGURE 3
MANEUVERING STABILITY
AH-56A USA S/N
SYMBOL
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL ANGLE ~DEG
TRIM AIRSPEED ~KCAS
CONFIGURATION
MANEUVER
19,600
299.6
5370
247
7.6
81
EXT STORES GEAR UP
LEFT WIND UP TURN
Determined from main rotor flap bending
MAIN ROTOR LIFT ~POUNDS
25,000
20,000
15,000
10,000
LATERAL STICK FORCE ~POUNDS
RT
LT
LATERAL STICK POSITION ~INCHES FROM CENTER
LONGITUDINAL STICK FORCE ~POUNDS
PULL
PUSH
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER
AFT
FWD
0
0.5
1.0
1.5
2.0
2.5
C.G. NORMAL ACCELERATION ~ G'S

FIGURE 14
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	18,800	300	3290	245	5.4	120	EXT STORES GEAR UP	LEFT WIND UP TURN



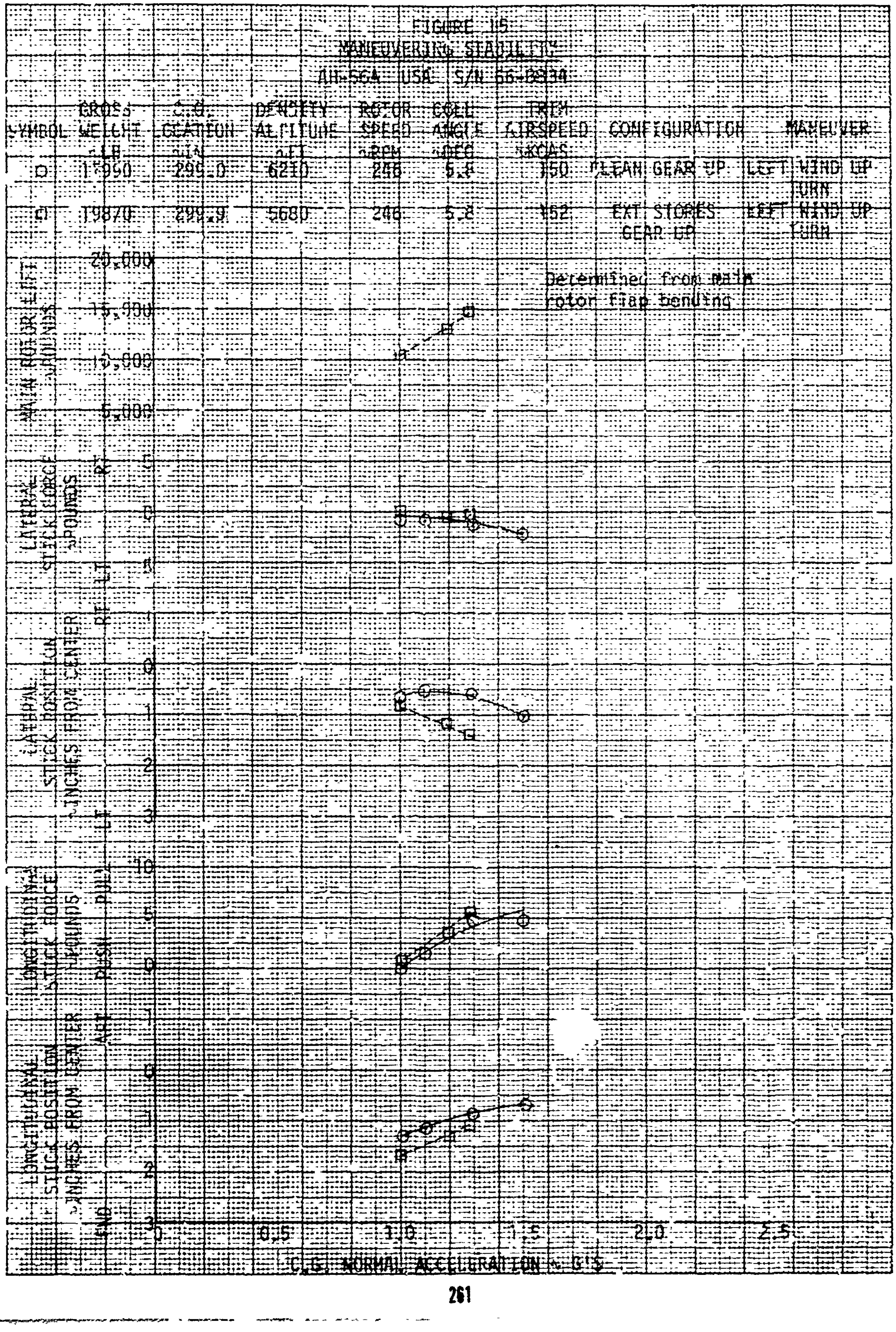


FIGURE 15
MANEUVERING STABILITY
AH-56A USA S/N 66-8834
SYMBOL
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL ANGLE ~DEG
TRIM AIRSPEED ~KCAS
CONFIGURATION
MANEUVER
1?950 299.0 6210 256 5.8 150 CLEAN GEAR UP LEFT WIND UP TURN
19870 299.9 5680 246 5.8 152 EXT STORES GEAR UP LEFT WIND UP TURN
Determined from main rotor flap bending
MAIN ROTOR LIFT ~POUNDS
20,000
15,000
10,000
5,000
LATERAL STICK FORCE ~POUNDS
RT
LT
LATERAL STICK POSITION ~INCHES FROM CENTER
RT
LT
LONGITUDINAL STICK FORCE ~POUNDS
PULL
PUSH
10
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER
AFT
FWD
0
0.5
1.0
1.5
2.0
2.5
C.G. NORMAL ACCELERATION ~ G'S

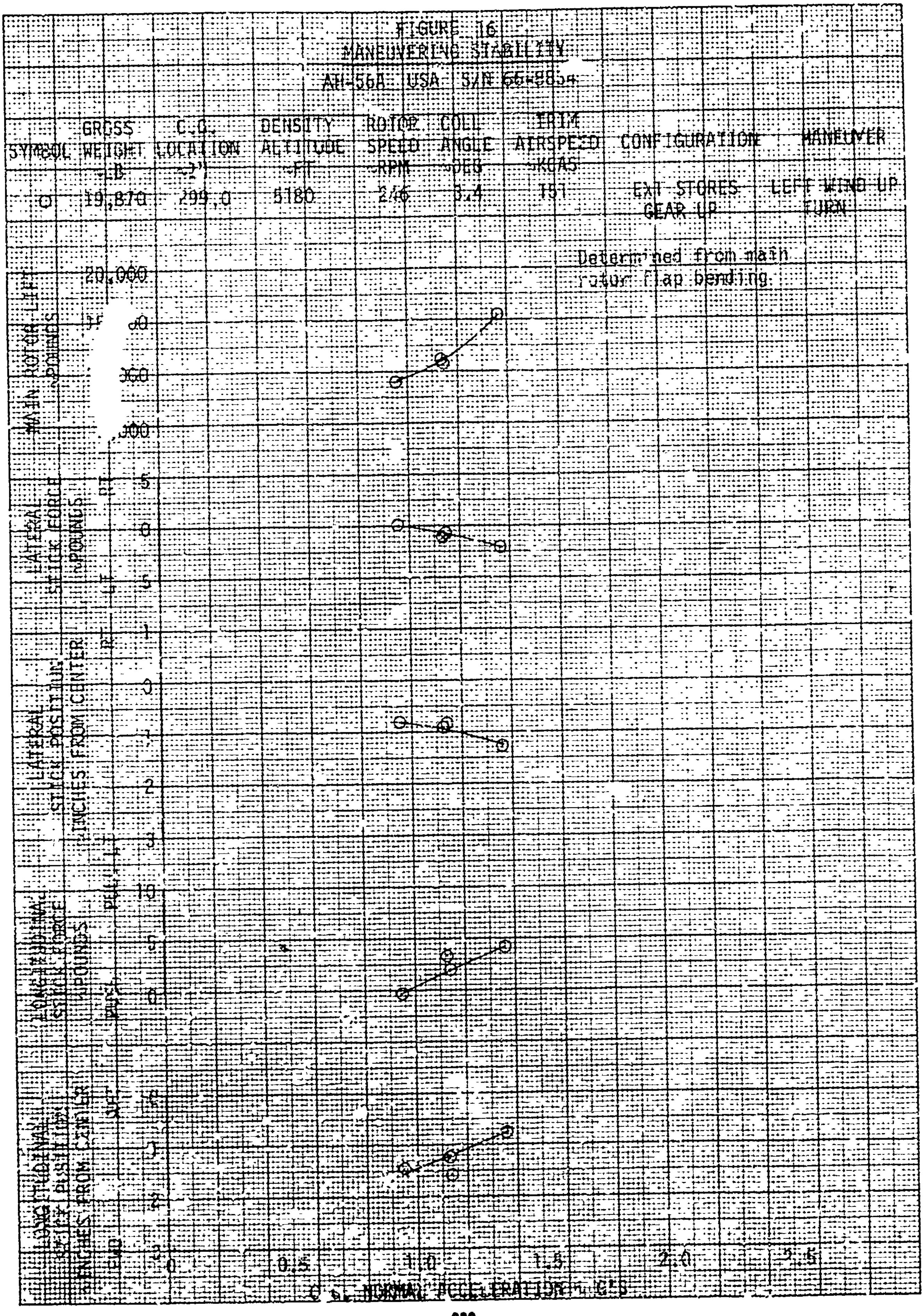


FIGURE 16
MANEUVERING STABILITY
AH-56A USA
SYMBOL
GROSS WEIGHT
C.G. LOCATION
DENSITY ALTITUDE
ROTOR SPEED
COLL ANGLE
TRIM AIRSPEED
CONFIGURATION
MANEUVER
19,870
299.0
5180
EXT STORES GEAR UP
LEFT WIND UP TURN
Determined from main rotor flap bending
20,000
MAIN ROTOR LIFT
LATERAL STICK FORCE
LATERAL STICK POSITION
INCHES FROM CENTER
LONGITUDINAL STICK FORCE
POUNDS
LONGITUDINAL STICK POSITION
INCHES FROM CENTER
0
0.5
1.0
1.5
2.0
2.5
NORMAL ACCELERATION

FIGURE 17
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	17,790	299.0	5210	246	5.3	180	CLEAN GEAR UP	LEFT WIND UP TURN

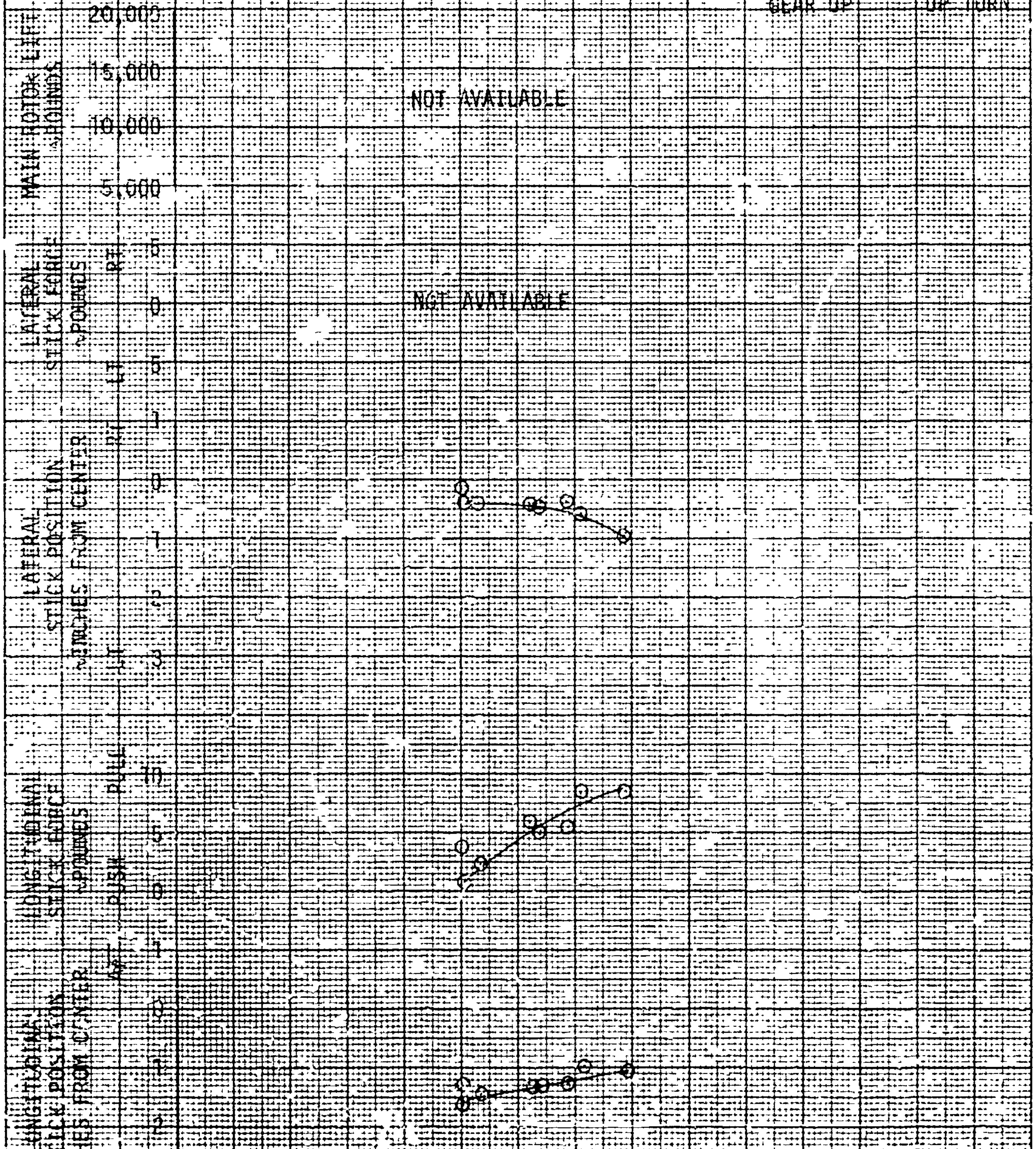


FIGURE 18
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

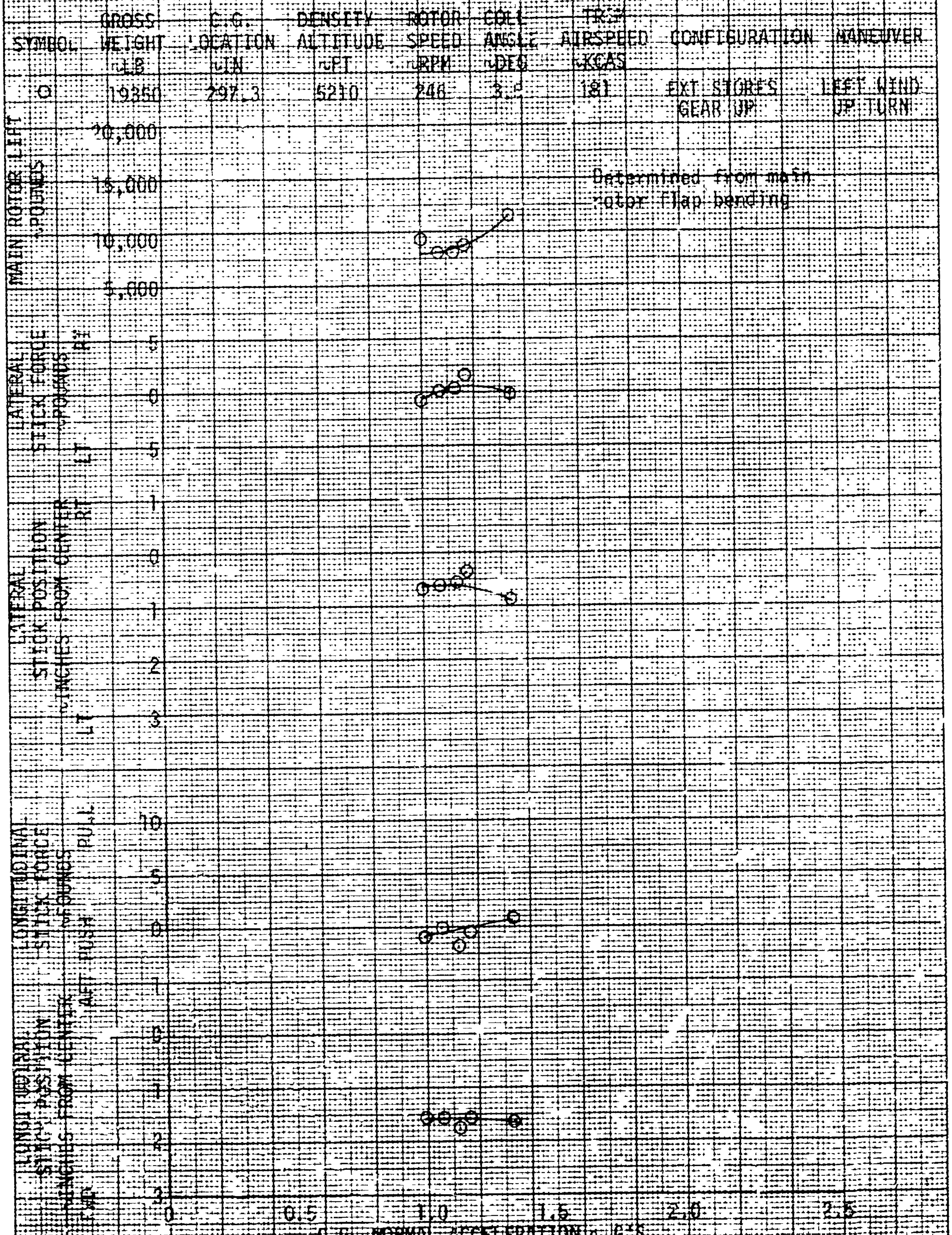


FIGURE 19
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	17,630	299.9	5130	246	6.5	181	CLEAN GEAR UP	RIGHT WIND UP TURN

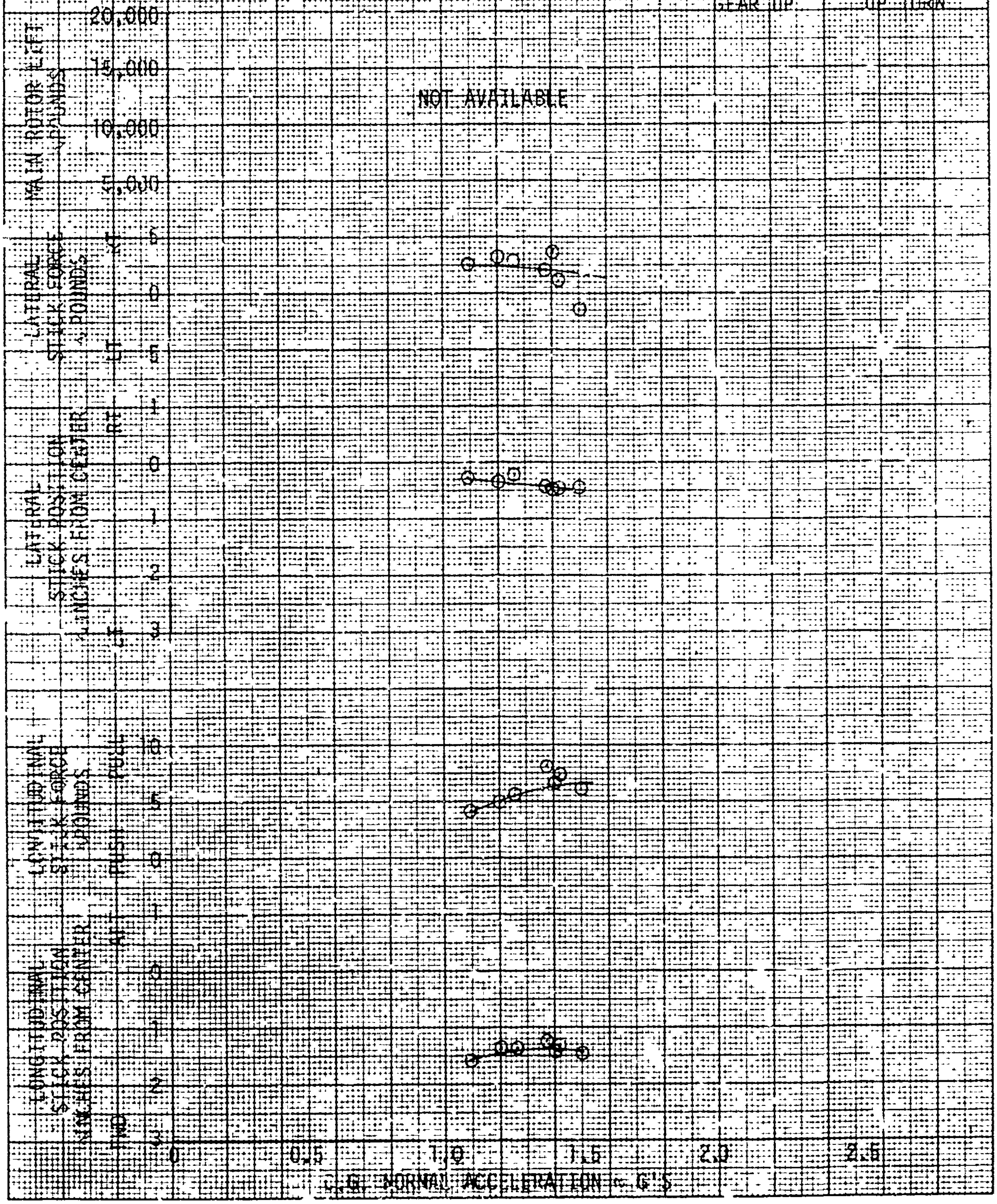


FIGURE 20
BLADE MOMENT STALL
AH-56A USA S/N 66-8834

LONGITUDINAL, PITCH & CG ACCEL
LATERAL & ROLL
PEDAL, YAW & SIDESLIP

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLLECTIVE ANGLE ~DEG	AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
17790	299.0	5210	246	5.3	180	CLEAN GEAR UP	LEFT WIND-UP TURN

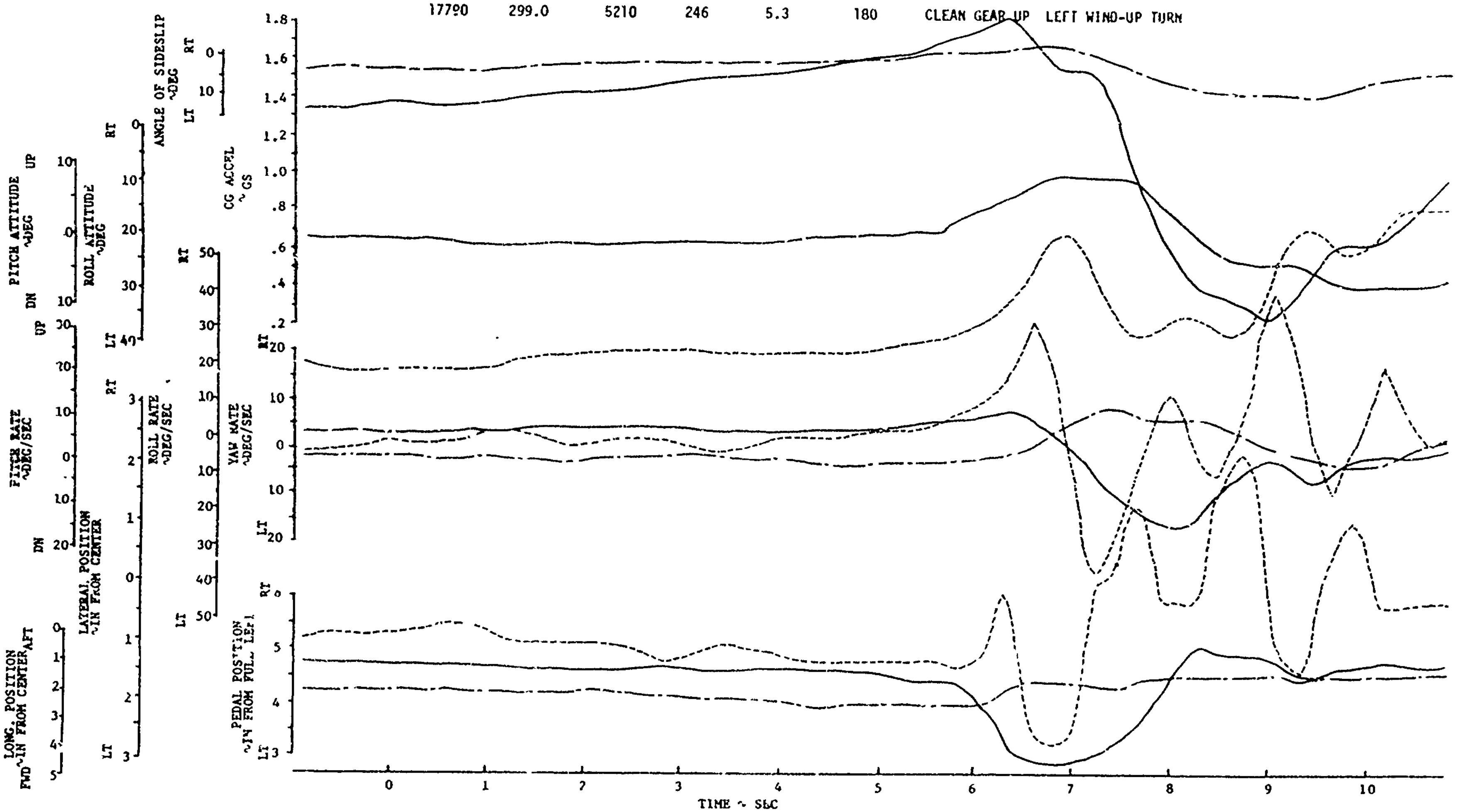


FIGURE 21
BLADE MOMENT STALL
AH-56A USA S/N 66-8334

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLLECTIVE ANGLE ~DEG	AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
19720	299.9	5240	247	5.3	178	EXT STORES GEAR UP	LEFT WIND-UP TURN

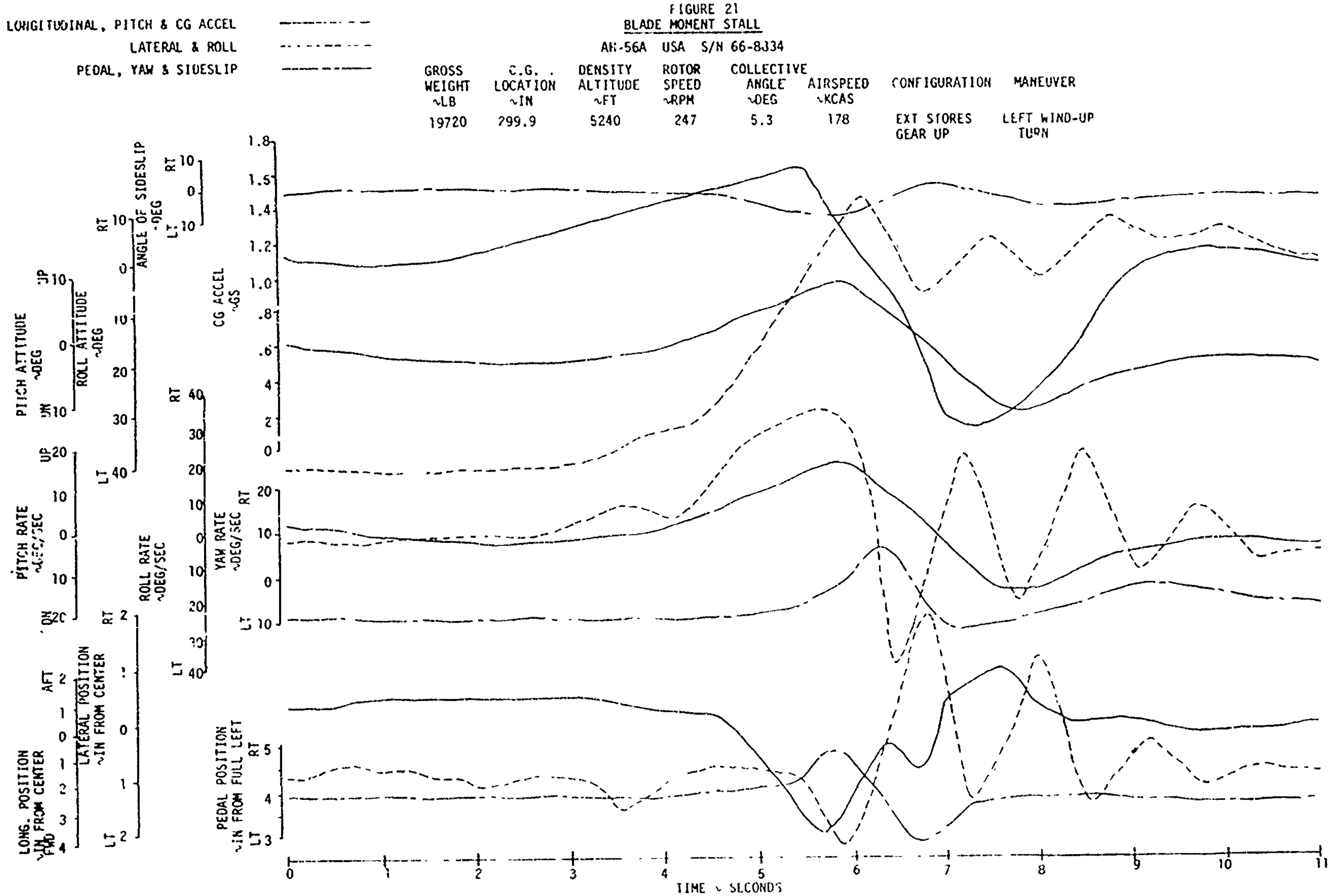


FIGURE 22
ACCELERATION THRU TRANSITION
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	ROTOR SPEED ~RPM	CONFIGURATION
18580	299.1	247	CLEAN GEAR DN

LONGITUDINAL, PITCH & COLLECTIVE ————
LATERAL & ROLL & PROP - - - - - - -
PEDAL, YAW & SIDESLIP — - —

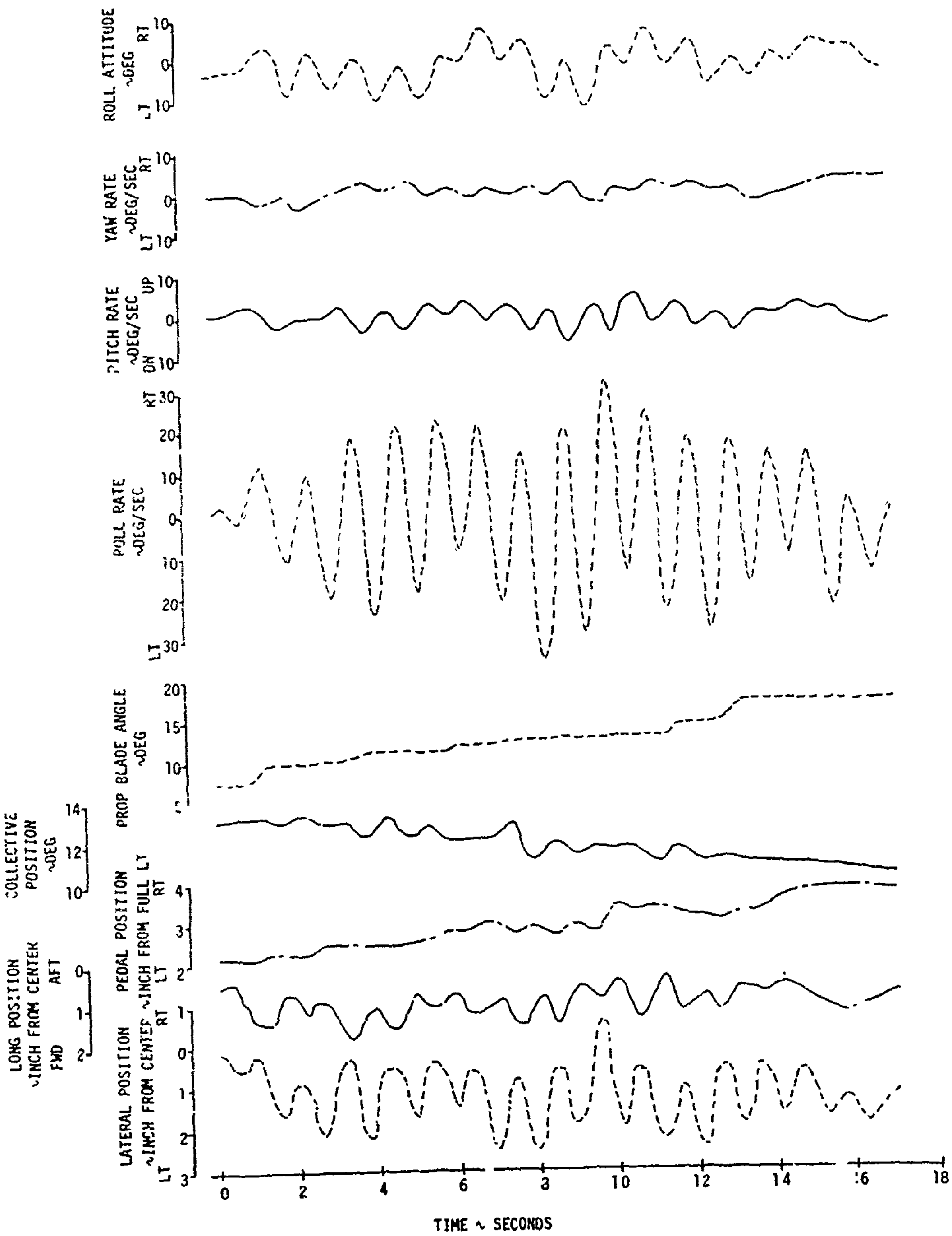
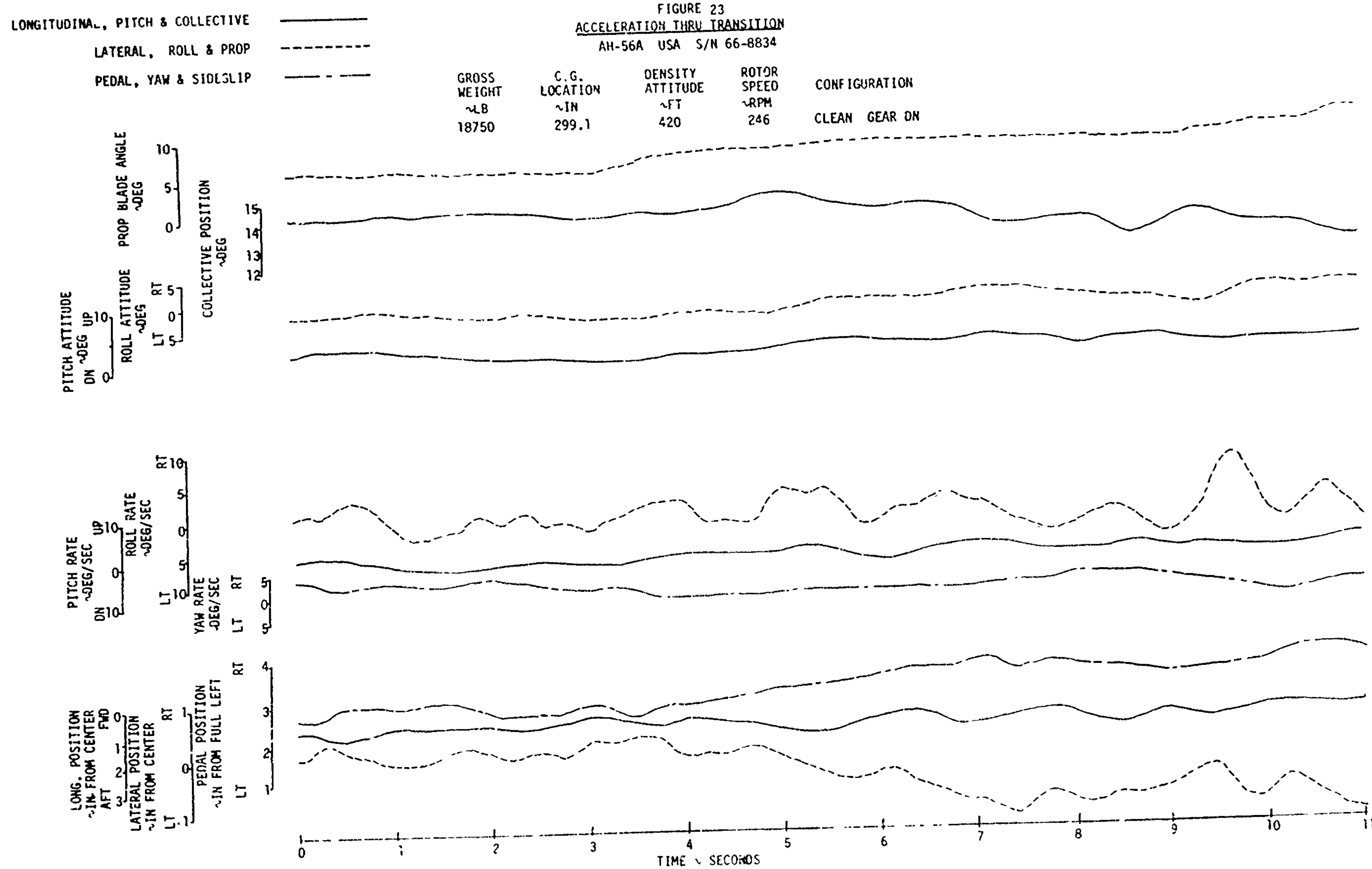


FIGURE 23
ACCELERATION THRU TRANSITION
AH-56A USA S/N 66-8834



LONGITUDINAL, PITCH & COLLECTIVE ————

LATERAL, ROLL & PPOP - - - - - - -

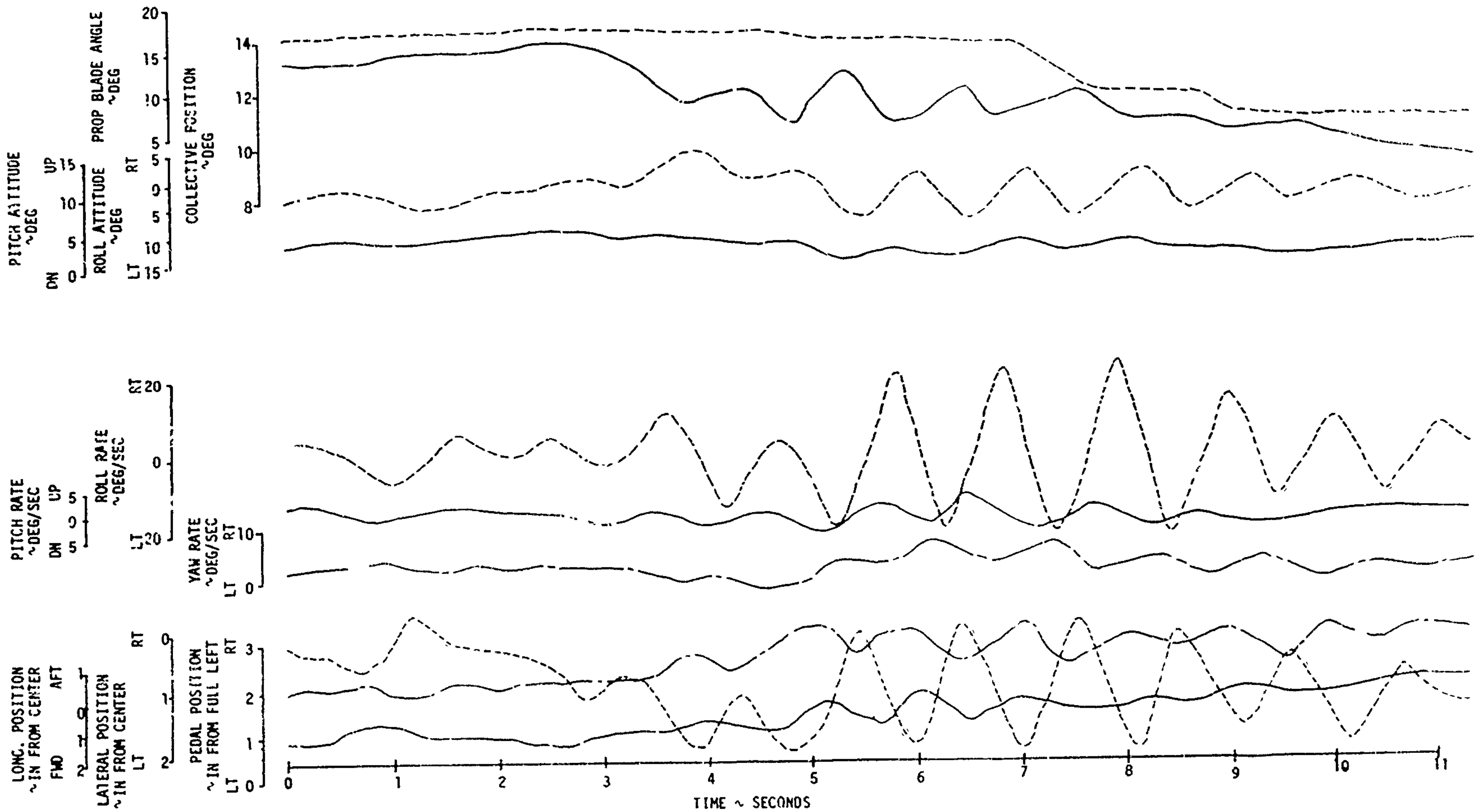
PEDAL, YAW & SIDESLIP —— - ——

FIGURE 24

ACCELERATION THRU TRANSITION

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION
18770	299.1	450	247	CL^EAN GEAR DN



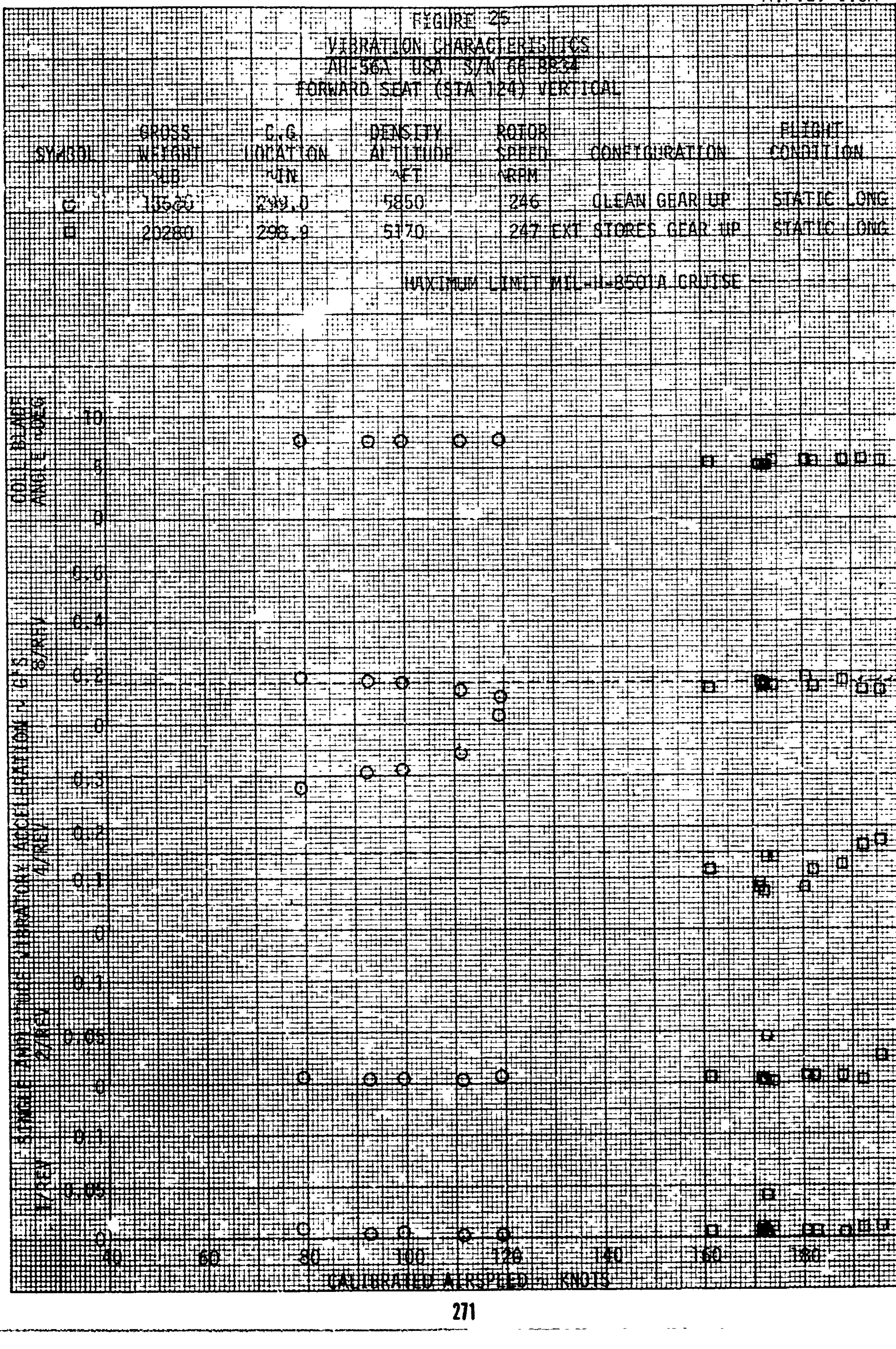


FIGURE 25
VIBRATION CHARACTERISTICS
FORWARD SEAT (STA 124) VERTICAL
SYMBOL
GROSS WEIGHT
C.G. LOCATION
DENSITY ALTITUDE
ROTOR SPEED
CONFIGURATION
FLIGHT CONDITION
5850
246
CLEAN GEAR UP
STATIC LONG
20280
298.9
5170
247
EXT STORES GEAR UP
STATIC LONG
MAXIMUM LIMIT MIL-H-8501A CRUISE
8/REV
4/REV
2/REV
1/REV
40
60
80
100
120
140
160
180
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 26
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
FORWARD SEAT (STA 124) LATERAL

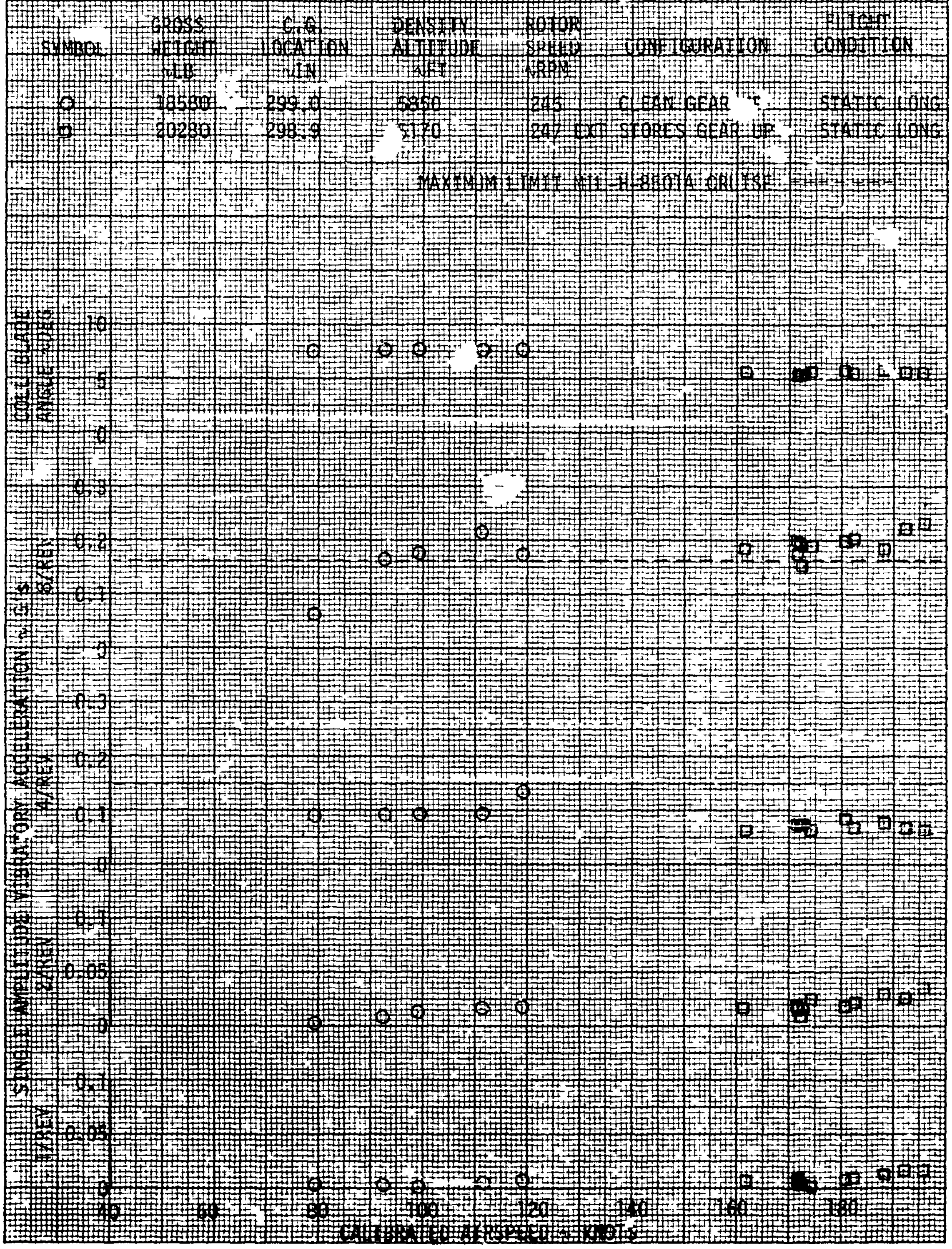


FIGURE 27
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
AFT SEAT (STA 170) VERTICAL

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
○	18580	299.0	5850	246	CLEAN GEAR UP	STATIC LONG
□	20280	298.9	5170	247	EXT STORES GEAR UP	STATIC LONG

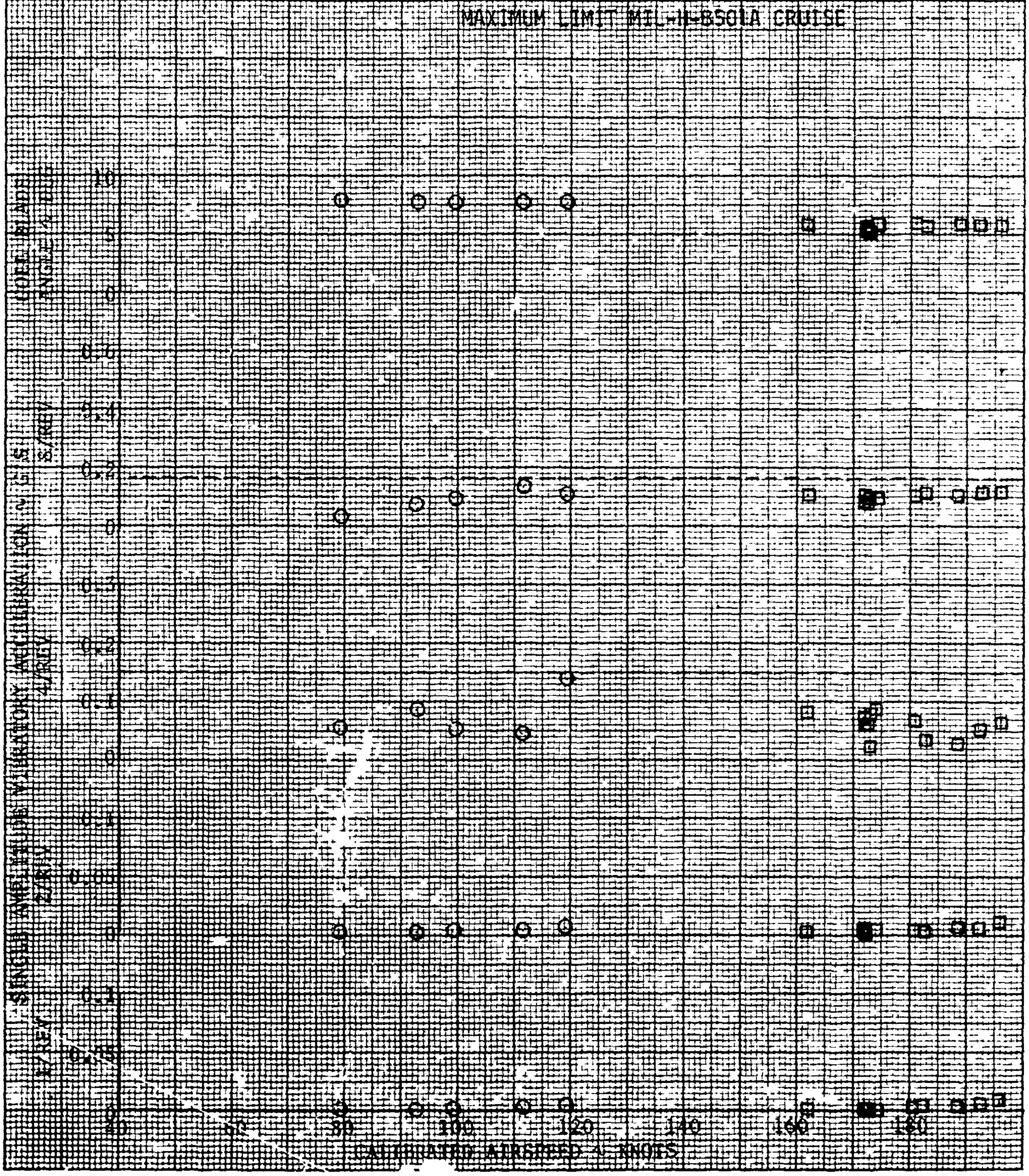


FIGURE 28
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
AFT SEAT (STA 170) LATERAL

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
○	18580	299.0	5850	246	CLEAN GEAR UP	STATIC LONG
□	20280	298.9	5170	247	EXT STORES GEAR UP	STATIC LONG

MAXIMUM LIMIT MIL-H-8501A CRUISE — — —

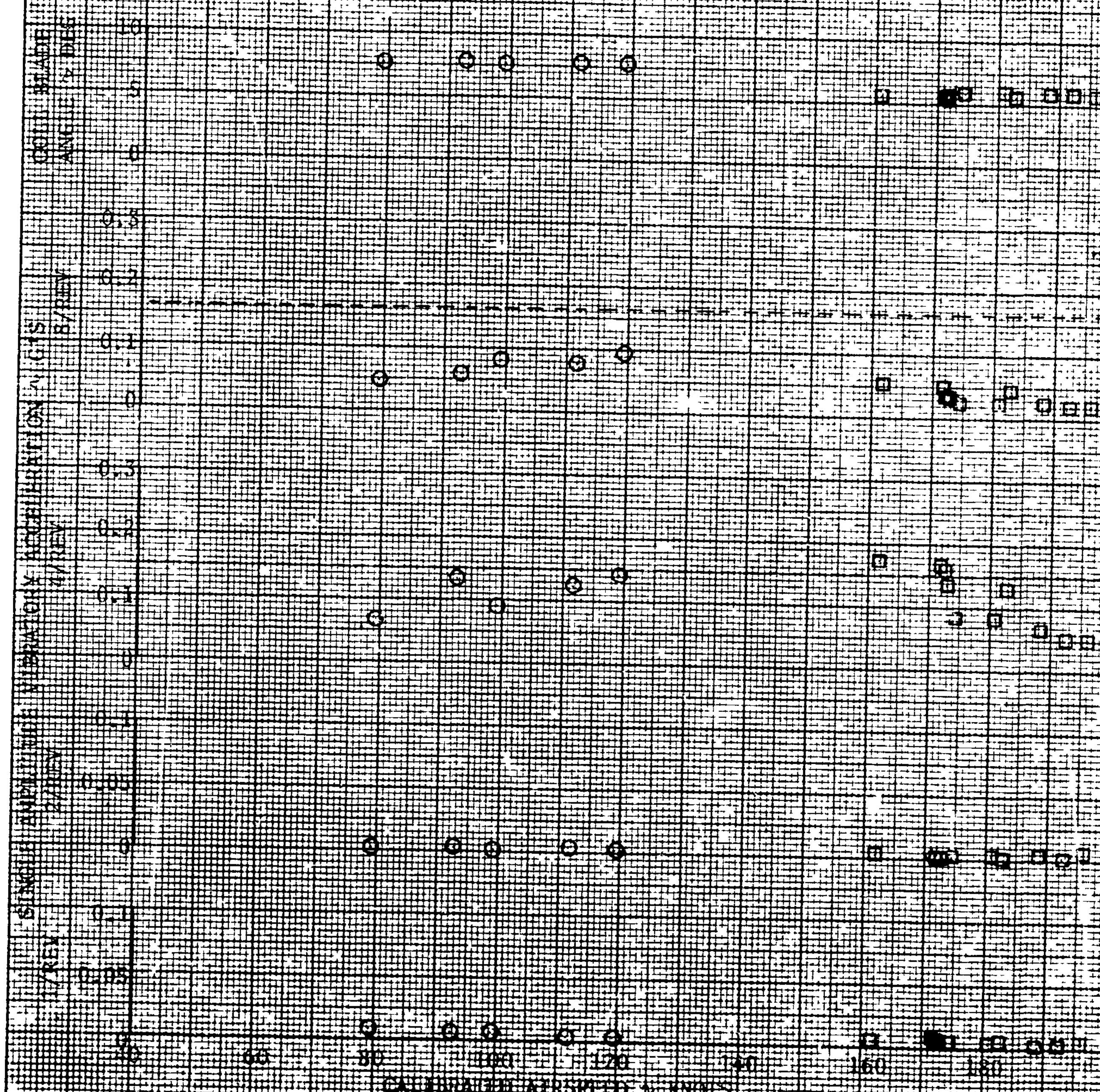


FIGURE 29
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
FORWARD SEAT (STA 124) VERTICAL

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
○	18820	298.9	4920	247	CLEAN GEAR UP	STATIC LAT. DIR 121 KTS

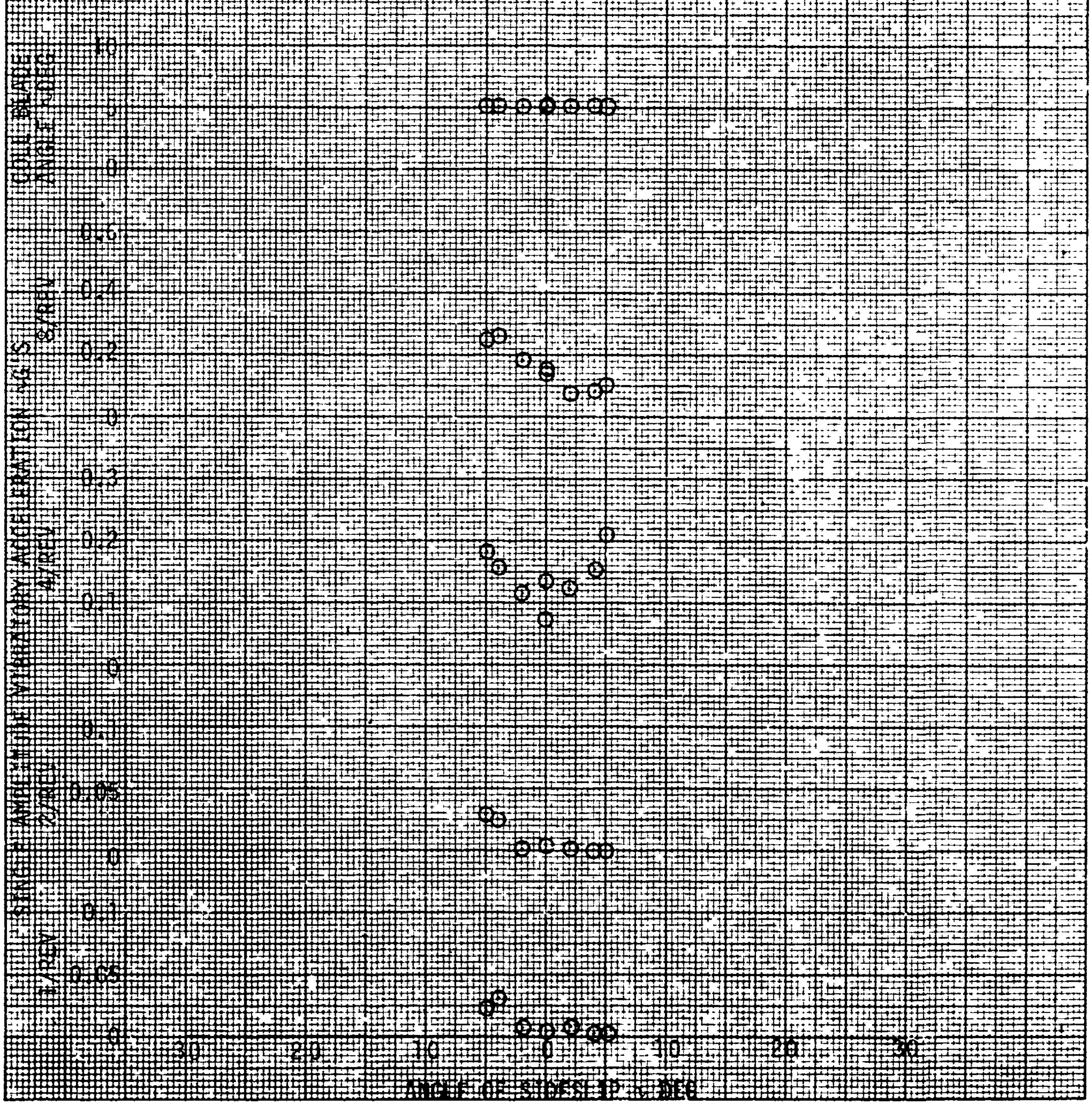


FIGURE 30

VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
FORWARD SEAT (STA 124) LATERAL

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
O	18820	298.9	1920	247	CLEAN GEAR UP	STATIC LAT-DIR 171 KTS

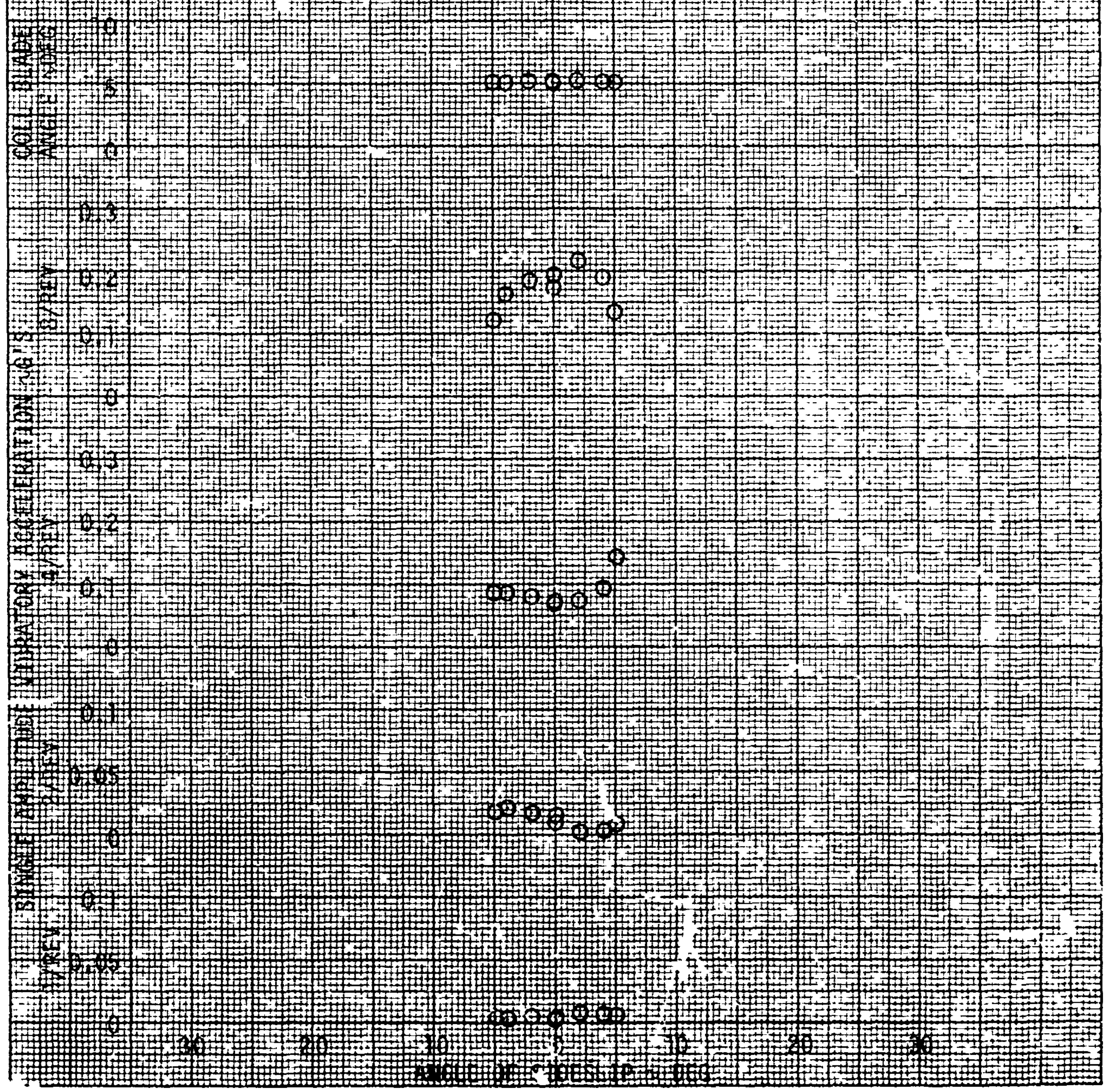


FIGURE 31
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
AFT SEAT (STA 176) VERTICAL

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
O	18820	298.9	4920	247	CLEAN GEAR UP	STATIC LAT.DIR 171 KTS

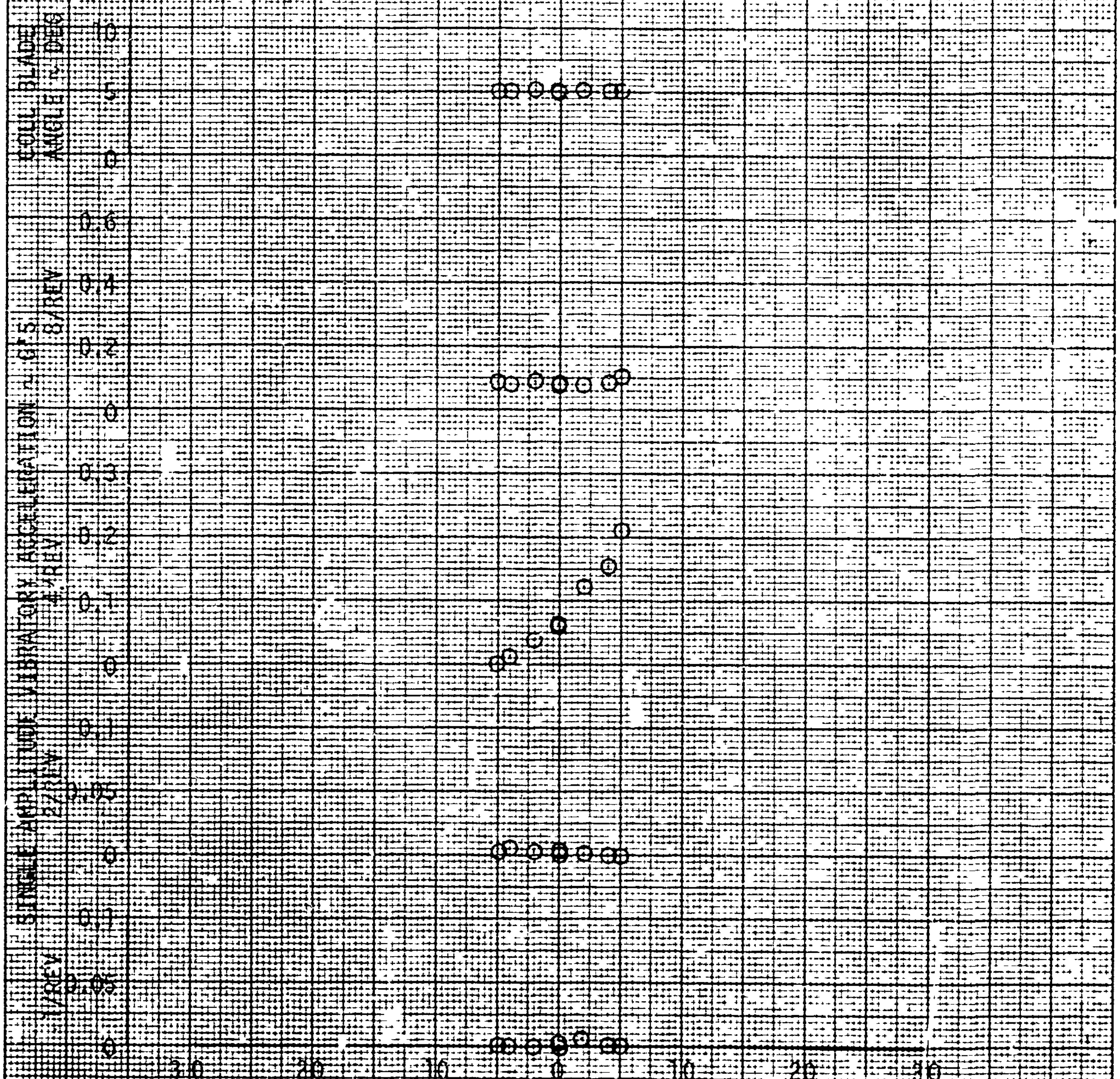
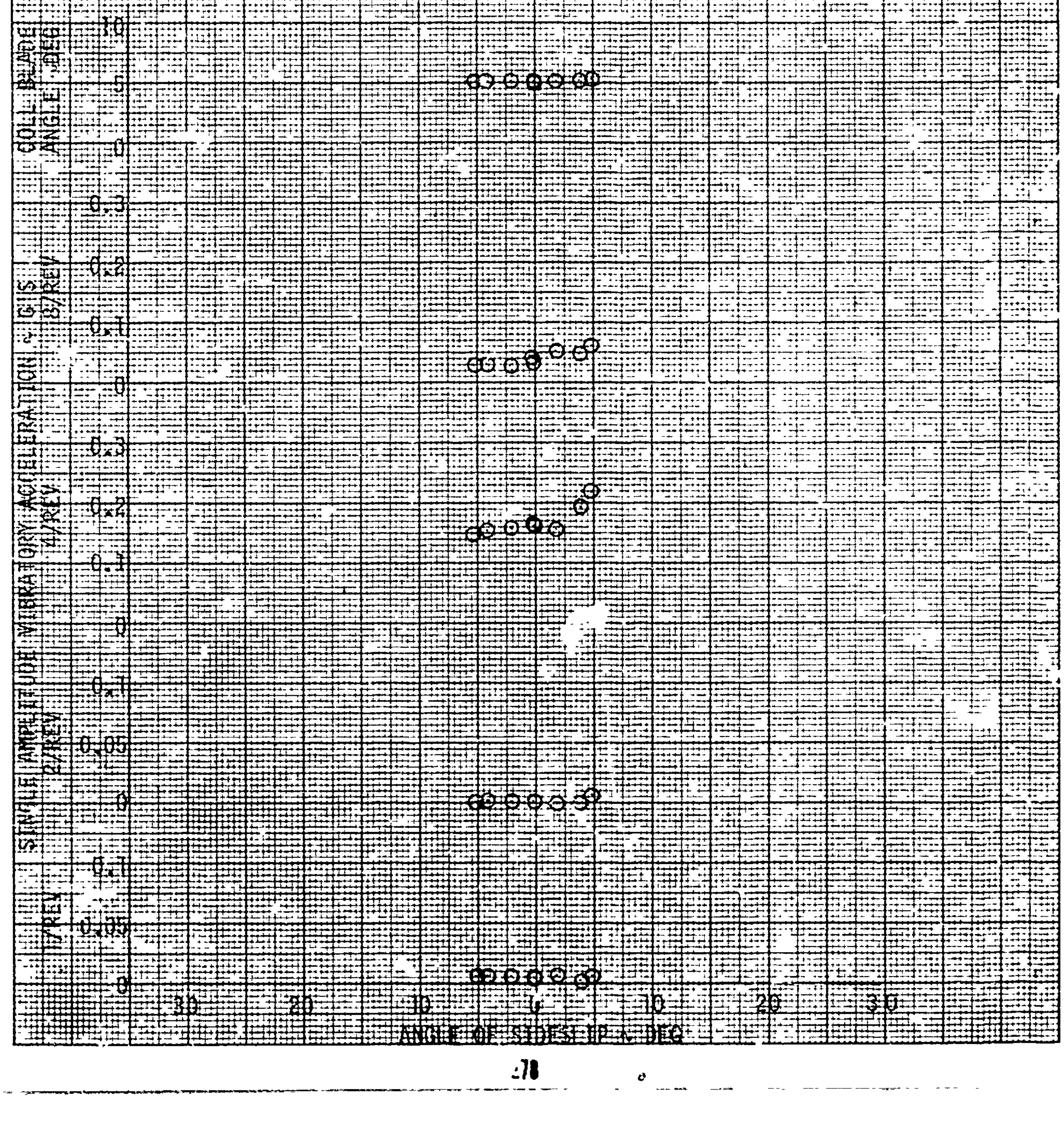


FIGURE 32
VIBRATION CHARACTERISTICS
AH-56A USA S/N 66-8834
AFT SEAT (STA 170) LATERAL

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
○	18820	298.9	4820	247	CLEAN GEAR UP	STATIC LAT.DIR 171 KTS



APPENDIX I. APE 1.3B TEST DATA

This appendix contains test data obtained during APE 1.3B. Prior to APE 1.3B, the lateral control system was modified by adding 16 and 3.. hertz filters to the roll compensator, activating the lift/roll decoupling feature, reducing the lateral control effectiveness from 60 percent (APE 1.1 and APE 1.2) to 45 percent of Phase III design control effectiveness, increasing the swashplate feedback, and decreasing the lateral f... play, breakout, and force gradients. Additionally, detuning weights were a... to the right wing and the aft collective control was detuned. All stability and control augmentation devices were activated for these tests. Tests were conducted at the following nominal conditions: 5000-foot density altitude, 100-percent rotor speed, 18,300- to 20,500-pound gross weight, station 2?2 (forward) to station 300 (aft) cg location, and in both the clean and external stores configuration. Trim airspeeds for each data point vary slightly from the presented airspeeds because of the difficulty and excessive time to obtain an exa... trim speed (para 27). This variation is not considered to have significantly aff... the data. The gross weight, center-of-gravity location, and density altitude st... on the st... ic longitudinal stability, static lateral-directional stability, m... ring stabil... and controllability figures ar... average values for the data pres...

INDEX

FIGURE 1
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
○	18230	299.1	4290	248	9.8	14	CLEAN(GEAR ON)

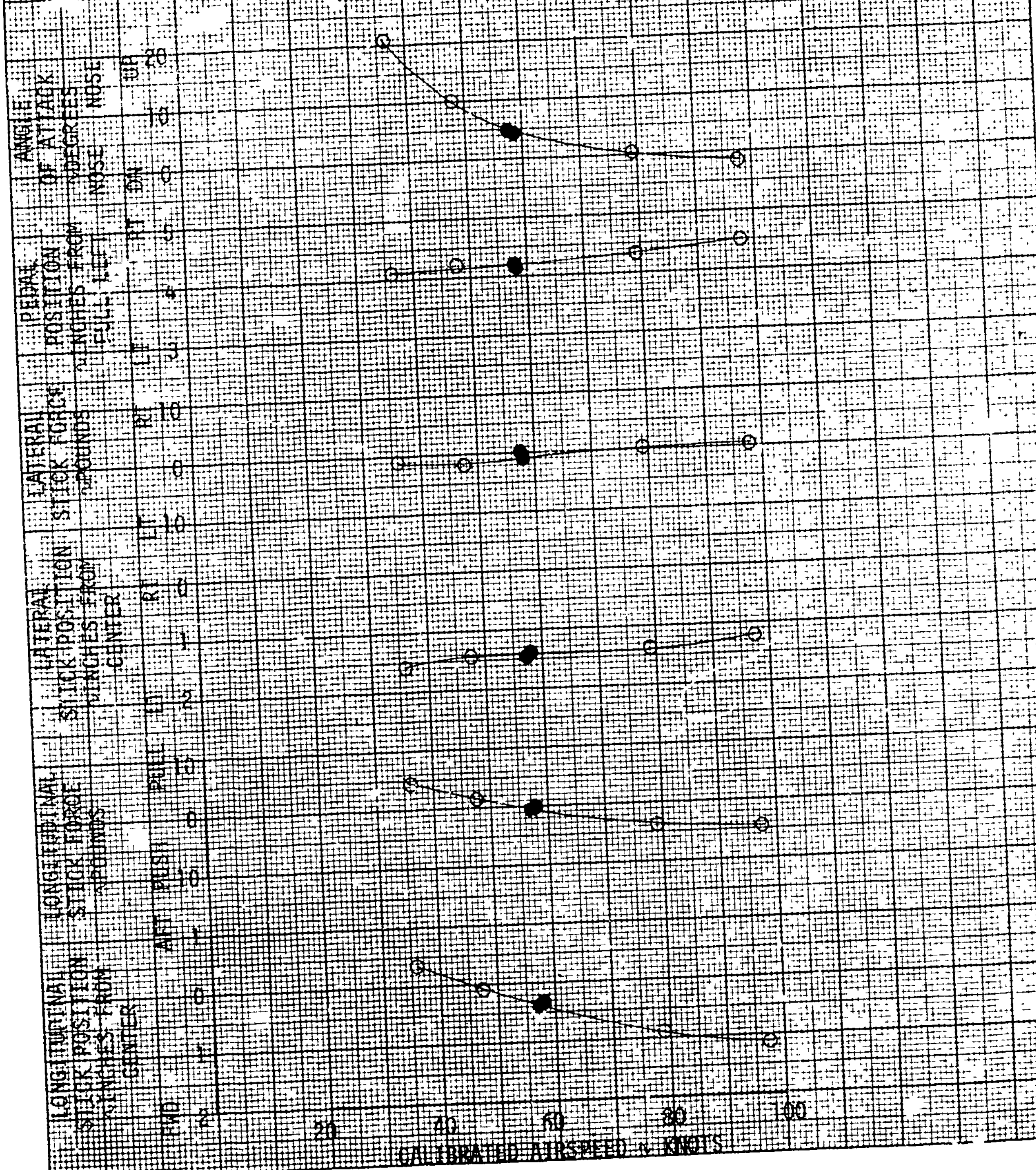


FIGURE 2
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
O	18110	300.0	5500	246	5.4	30.0	CLEAN (GEAR UP)

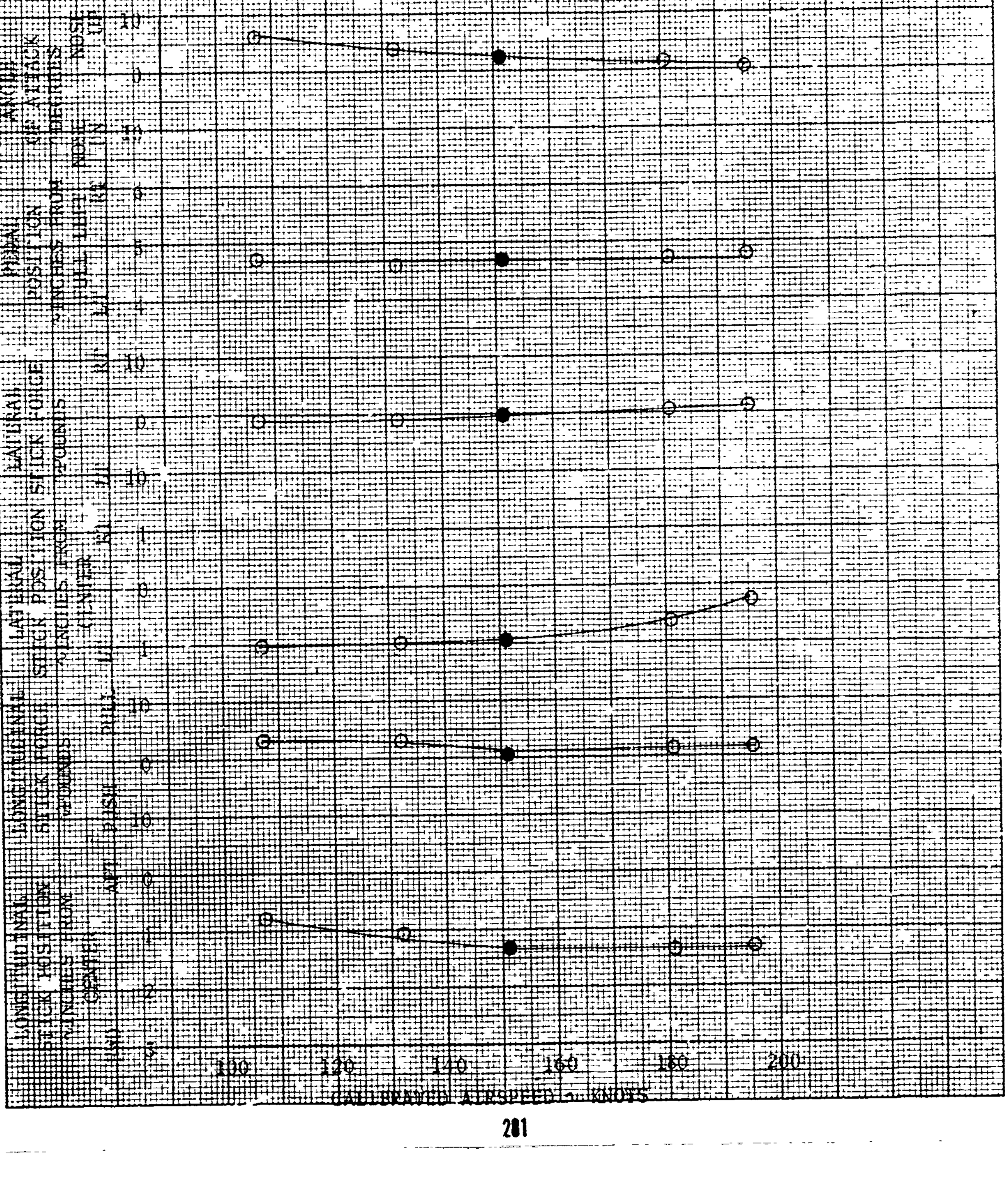


FIGURE 3
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
○	20030	299.8	4610	247	5.4	28	EXT STORE (GEAR UP)

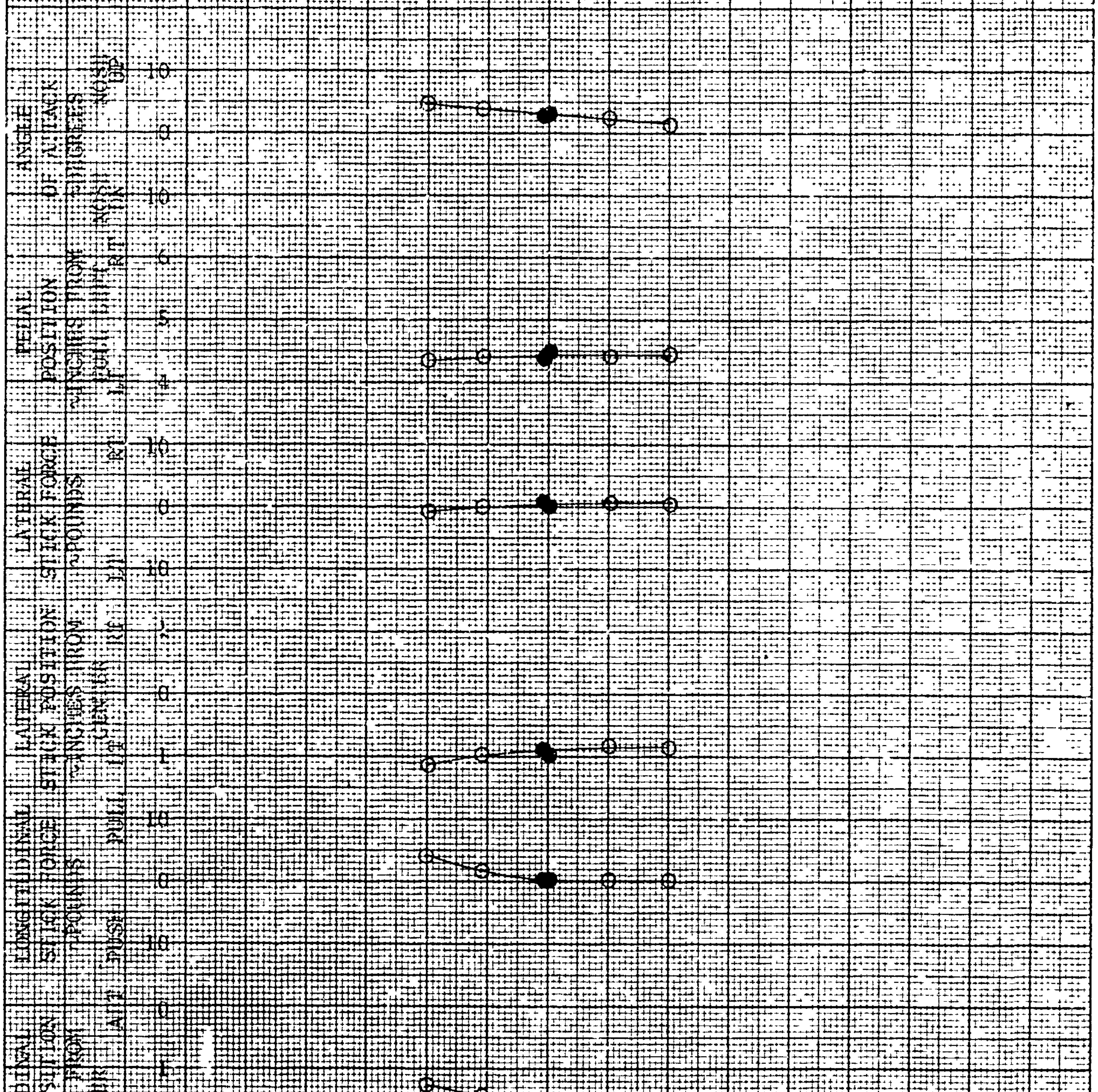


FIGURE 4
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	C.W. ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
O	13890	292.3	3840	248	3.7	31	CLEAN(GEAR UP

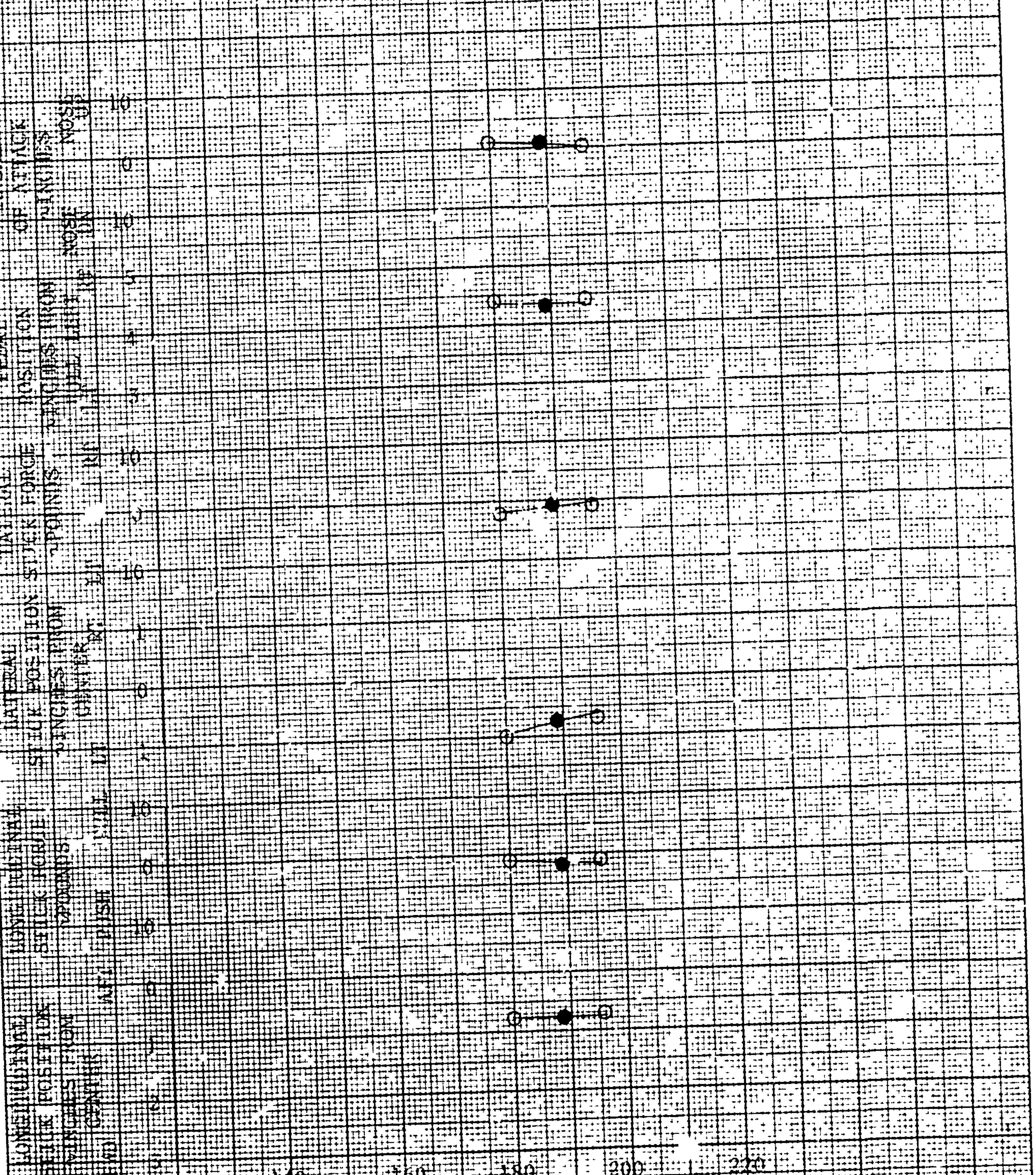


FIGURE 5

SUMMARY STATIC LATERAL-DIRECTIONAL STABILITY

AH-1GA USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION
○ (flagged)	17980	298.9	6530	244	CLEAN GEAR ON APE I.3.A
□ (flagged)	18070	299.1	4700	248	CLEAN GEAR ON APE I.3.B
○	18230	299.9	5390	246	CLEAN GEAR UP APE I.3.A
□	18530	300.0	5260	247	CLEAN GEAR UP APE I.3.B
●	19740	299.9	5440	246	EXT STORES GEAR UP APE I.3.A

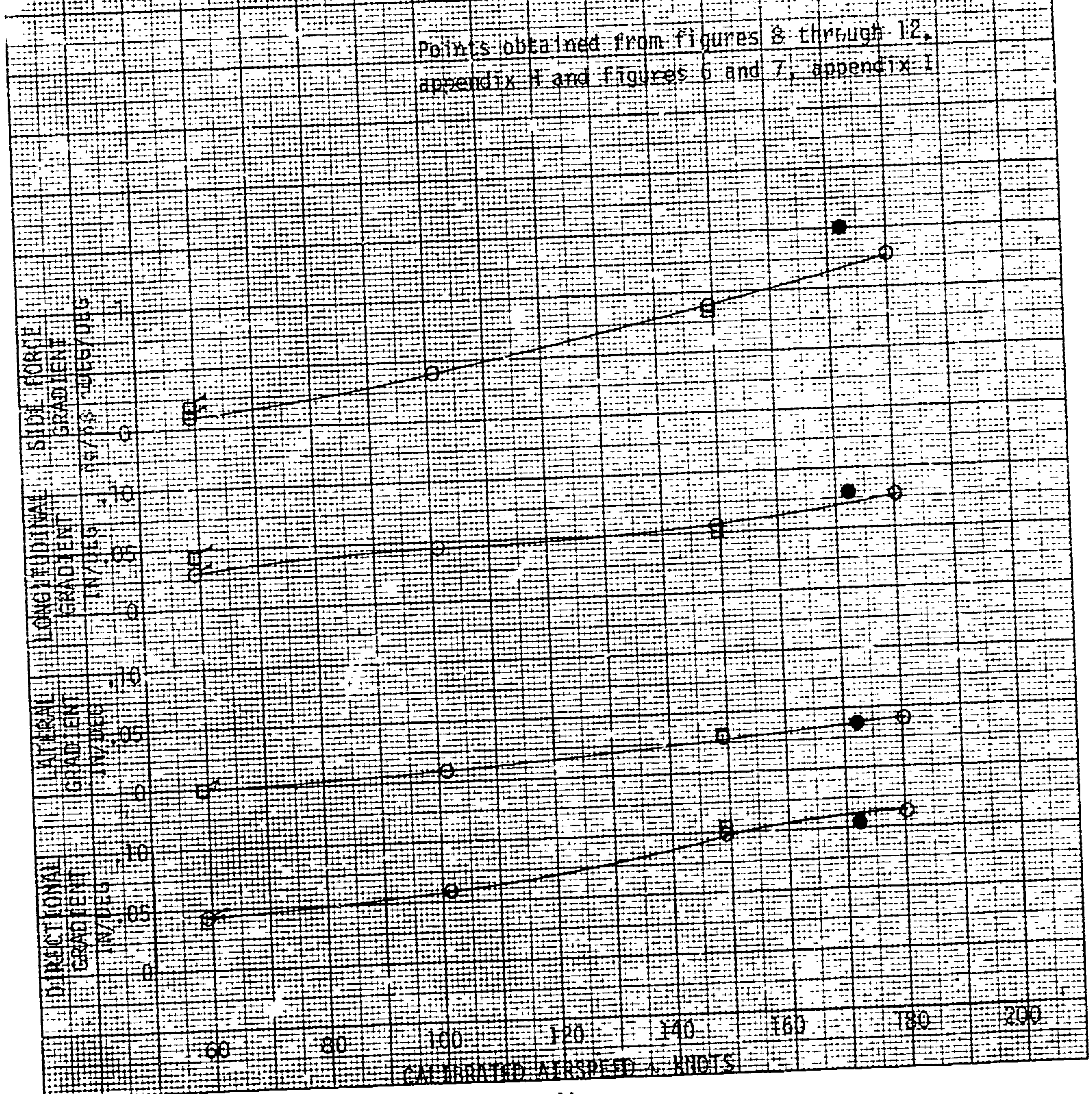


FIGURE 6
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION
○	18070	299.1	4700	248	7.8	15	59	CLEAN(GEAR UN)

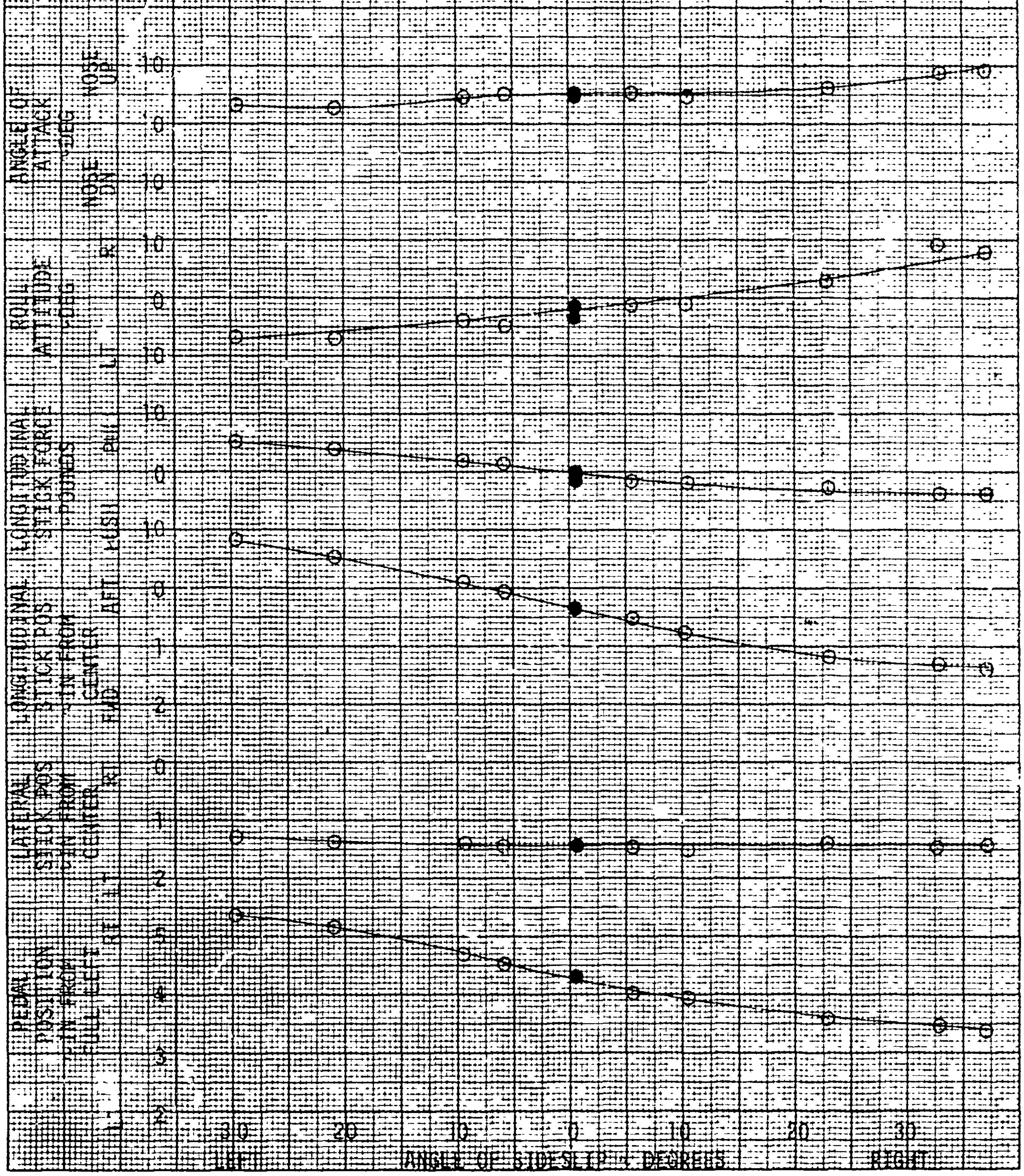


FIGURE 7
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	PROP ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION
○	18530	300.0	5260	247	6.0	28.4	149	CLEAN (GEAR UP)

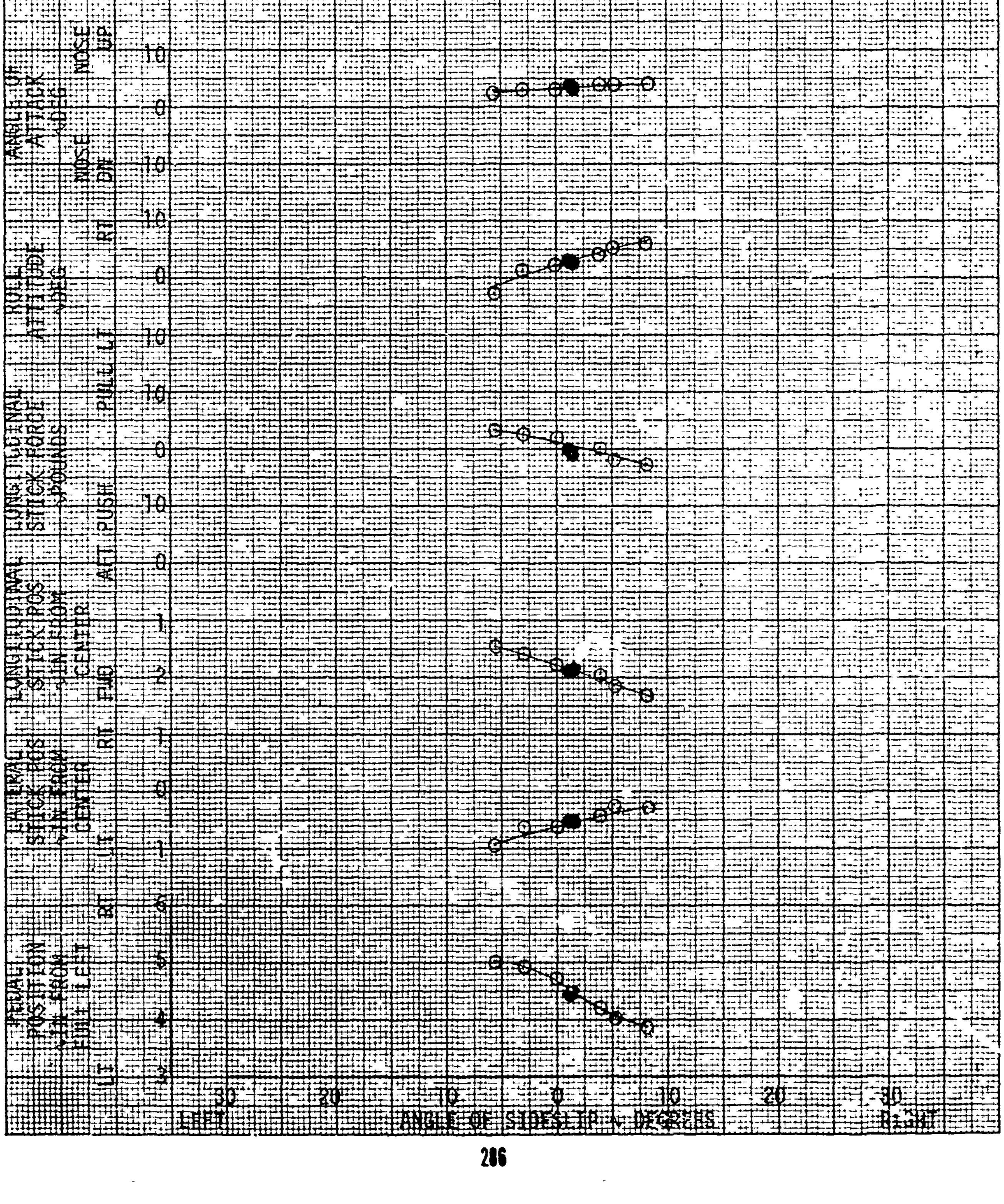


FIGURE 8
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	18300	301.2	4840	248	7.6	80	CLEAN GEAR UP	LEFT WIND UP TURN

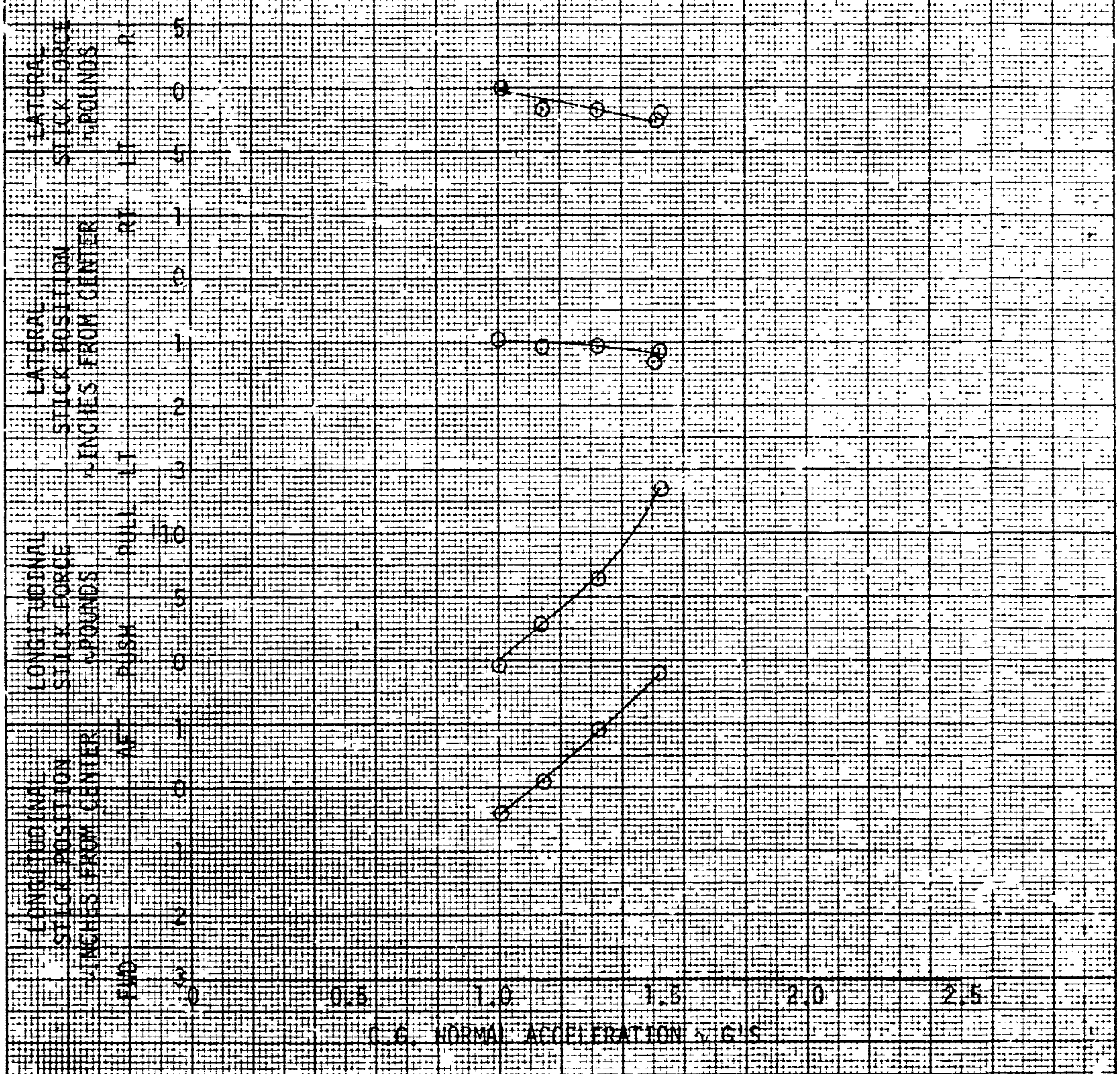


FIGURE 9
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
O	18420	300.1	4630	242	5.8	119	CLEAN GEAR UP	LEFT WIND UP TURN

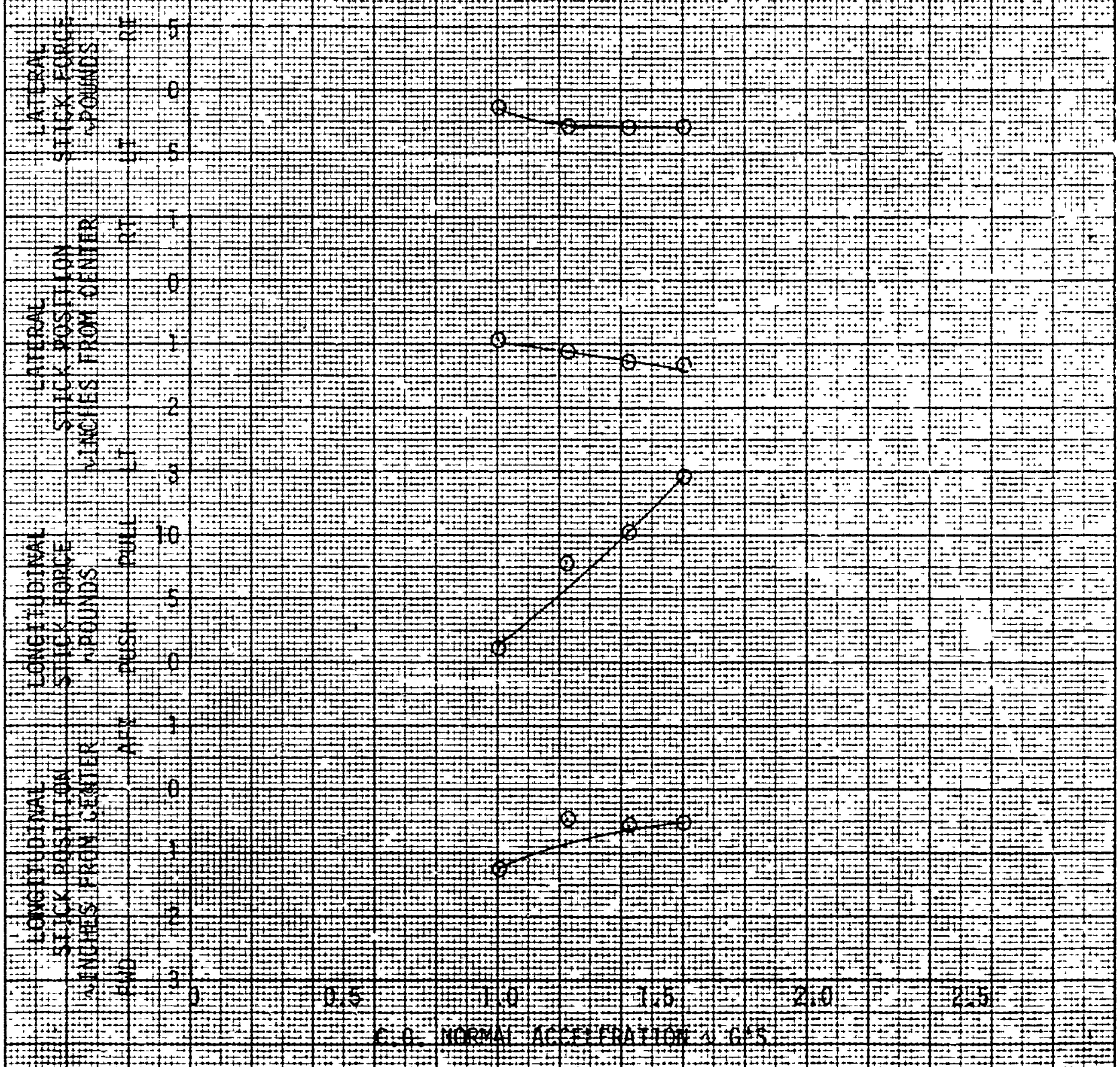


FIGURE 10
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	18580	300.2	4570	247	8.8	151	CLEAN GEAR UP	LEFT WIND UP TURN

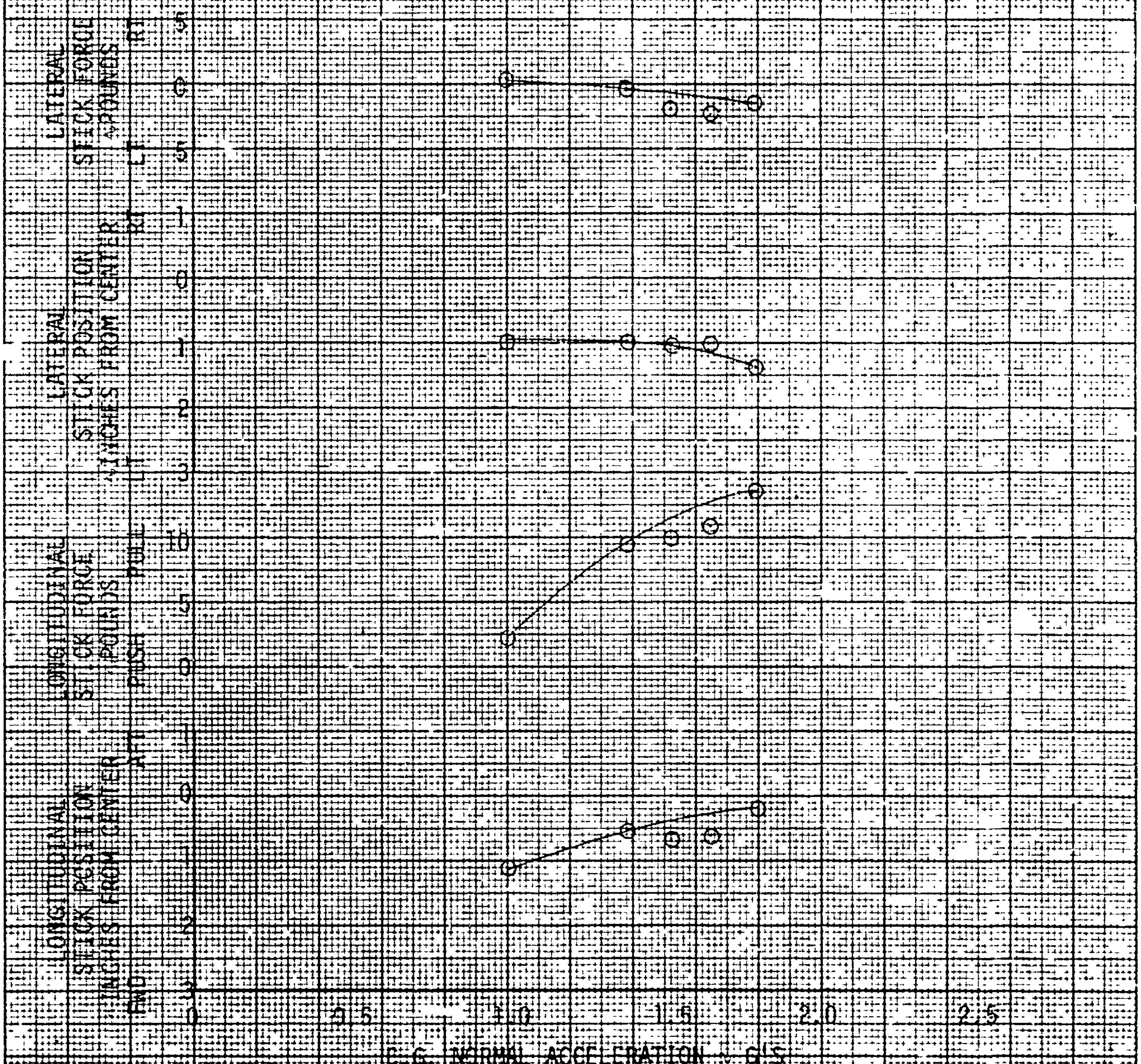


FIGURE 11
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	18230	300.1	5400	247	5.8	150	CLEAN GEAR UP	LEFT WIND UP TURN

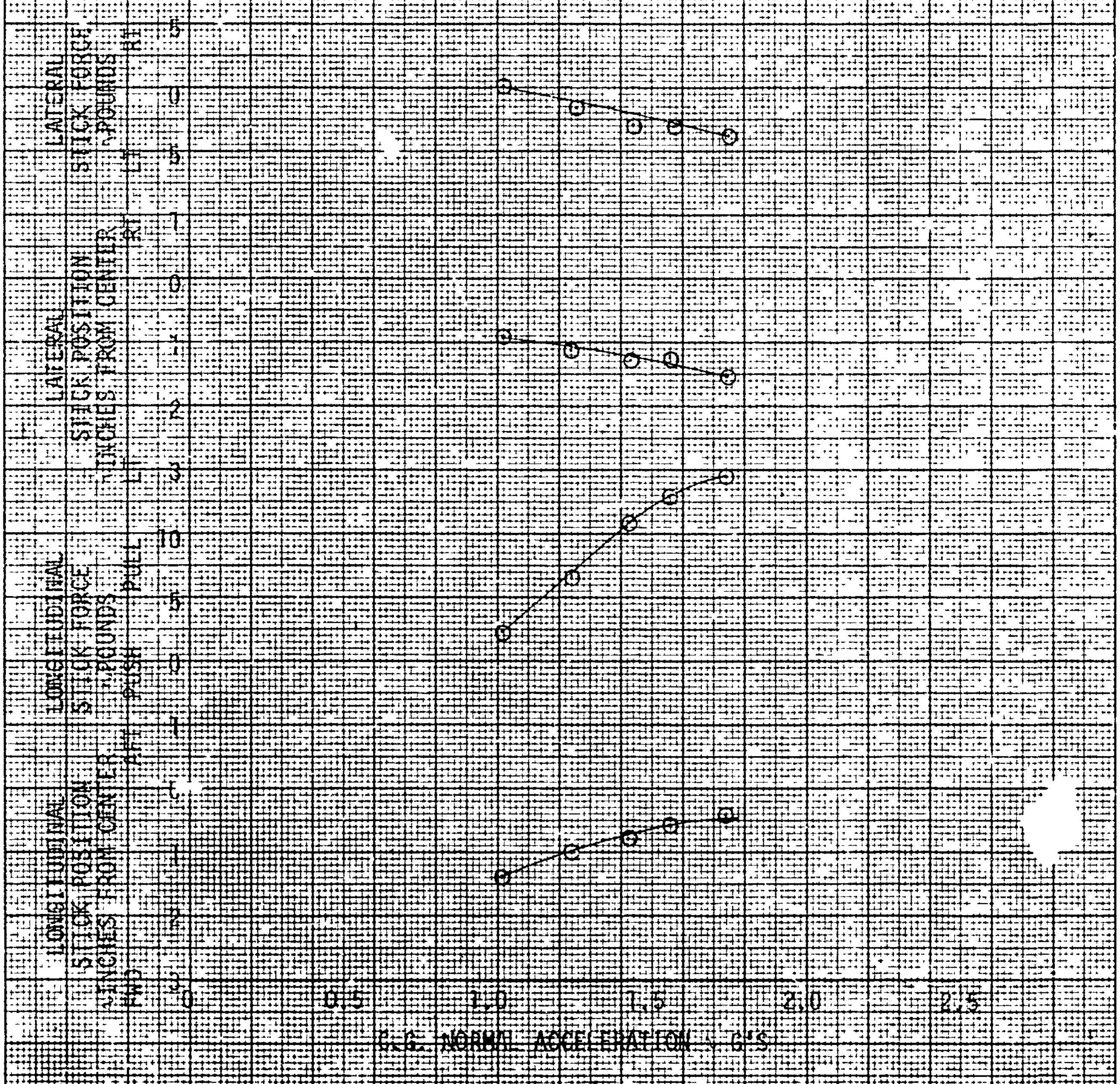


FIGURE 12
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	18250	293.1	5500	250	5.8	149	CLEAN GEAR UP	LEFT WIND UP TURN

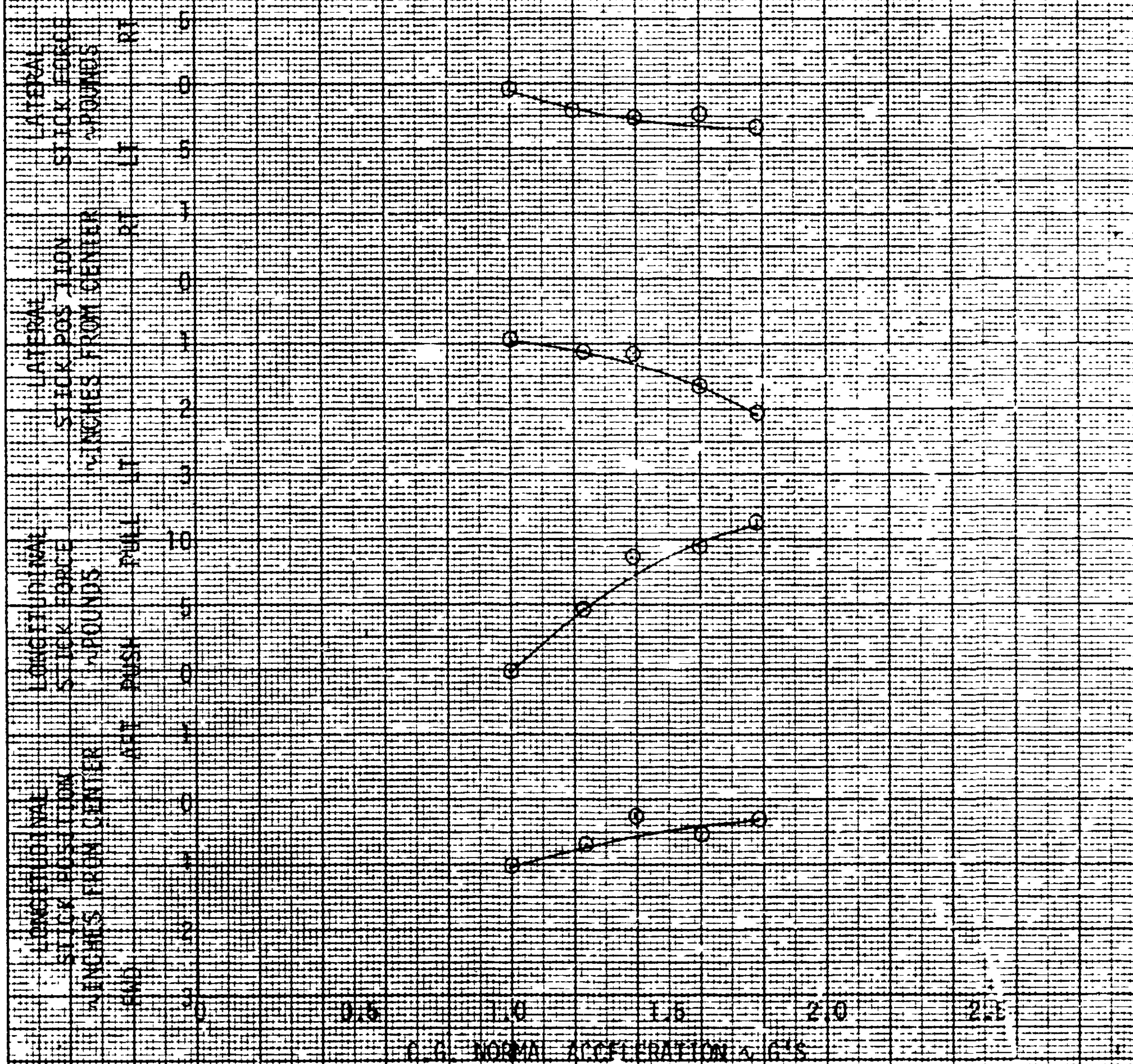


FIGURE 13
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
□	17870	300.0	5500	247	3.8	152	CLEAN GEAR UP	RIGHT WIND UP TURN

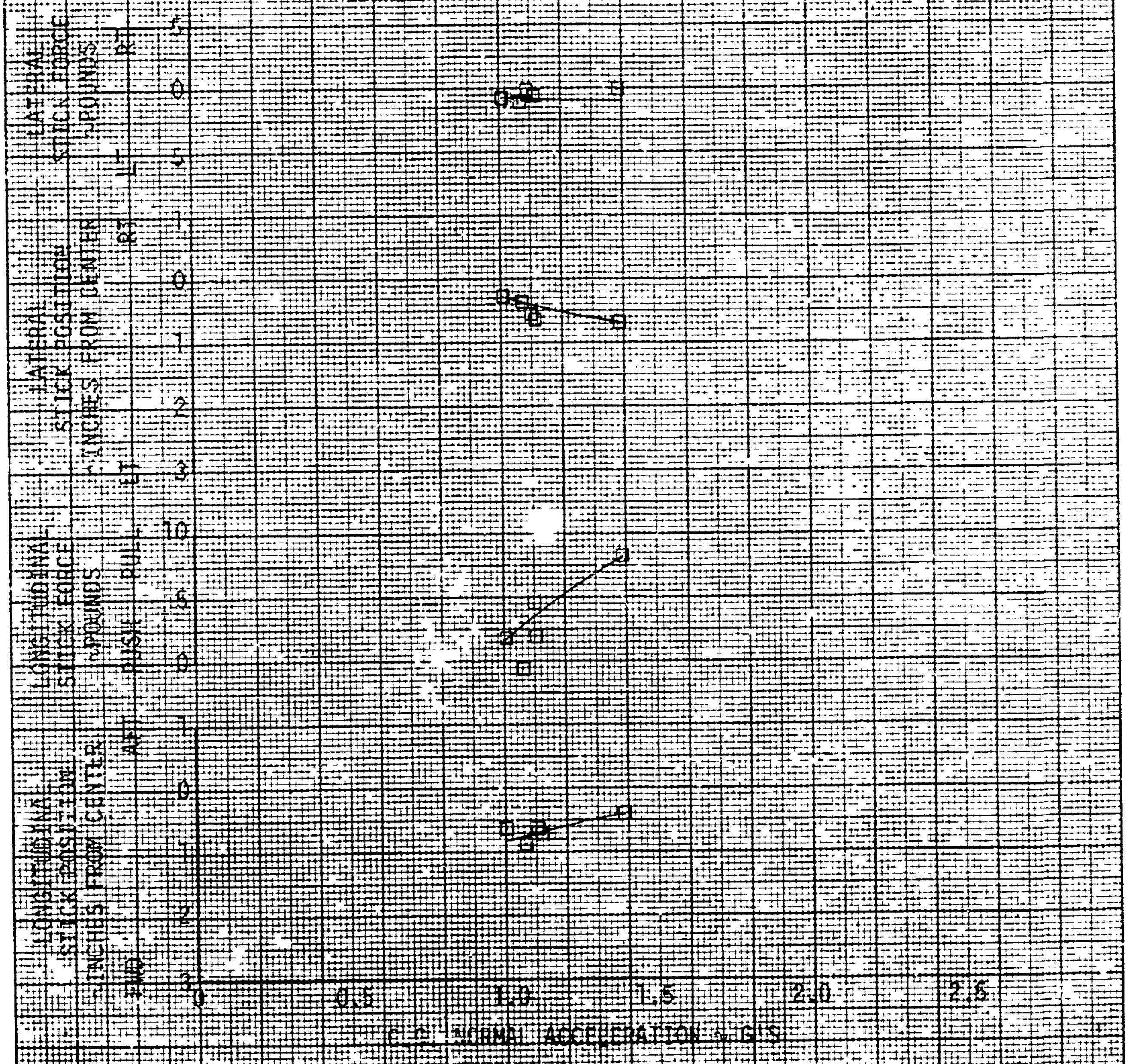


FIGURE 14
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	20210	299.9	4100	248	5.5	151	EXT STORES GEAR UP	LEFT WIND UP TURN

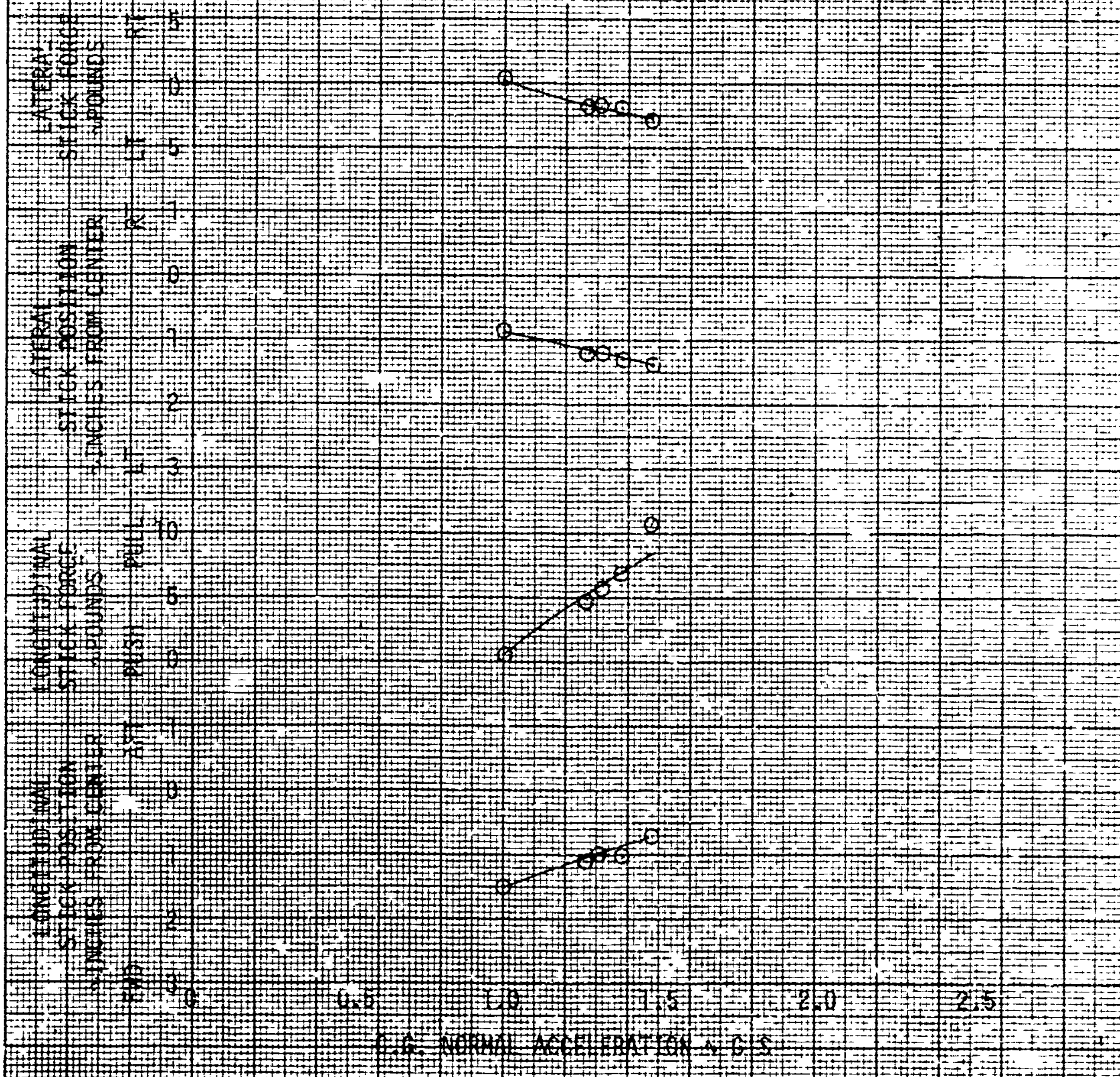


FIGURE 15
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
□	18410	300.2	3750	247	3.8	182	CLEAN GEAR UP	LEFT WIND UP TURN
○	18340	300.1	5890	248	5.8	182	CLEAN GEAR UP	LEFT WIND UP TURN

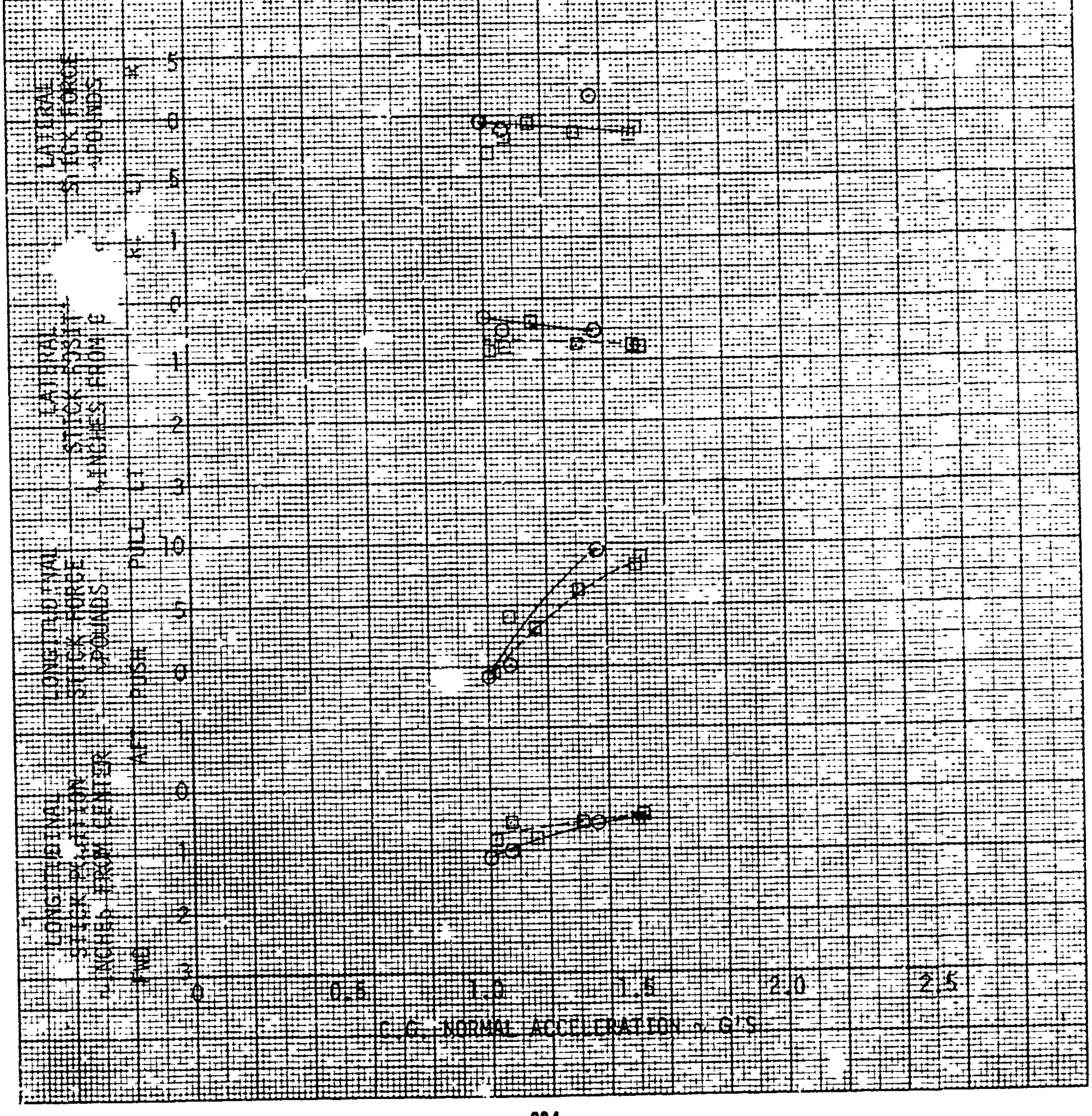


FIGURE 16

MANEUVERING STABILITY

AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
O	18040	283.0	4660	248	5.5	181	CLEAN GEAR UP	LEFT WIND UP TURN

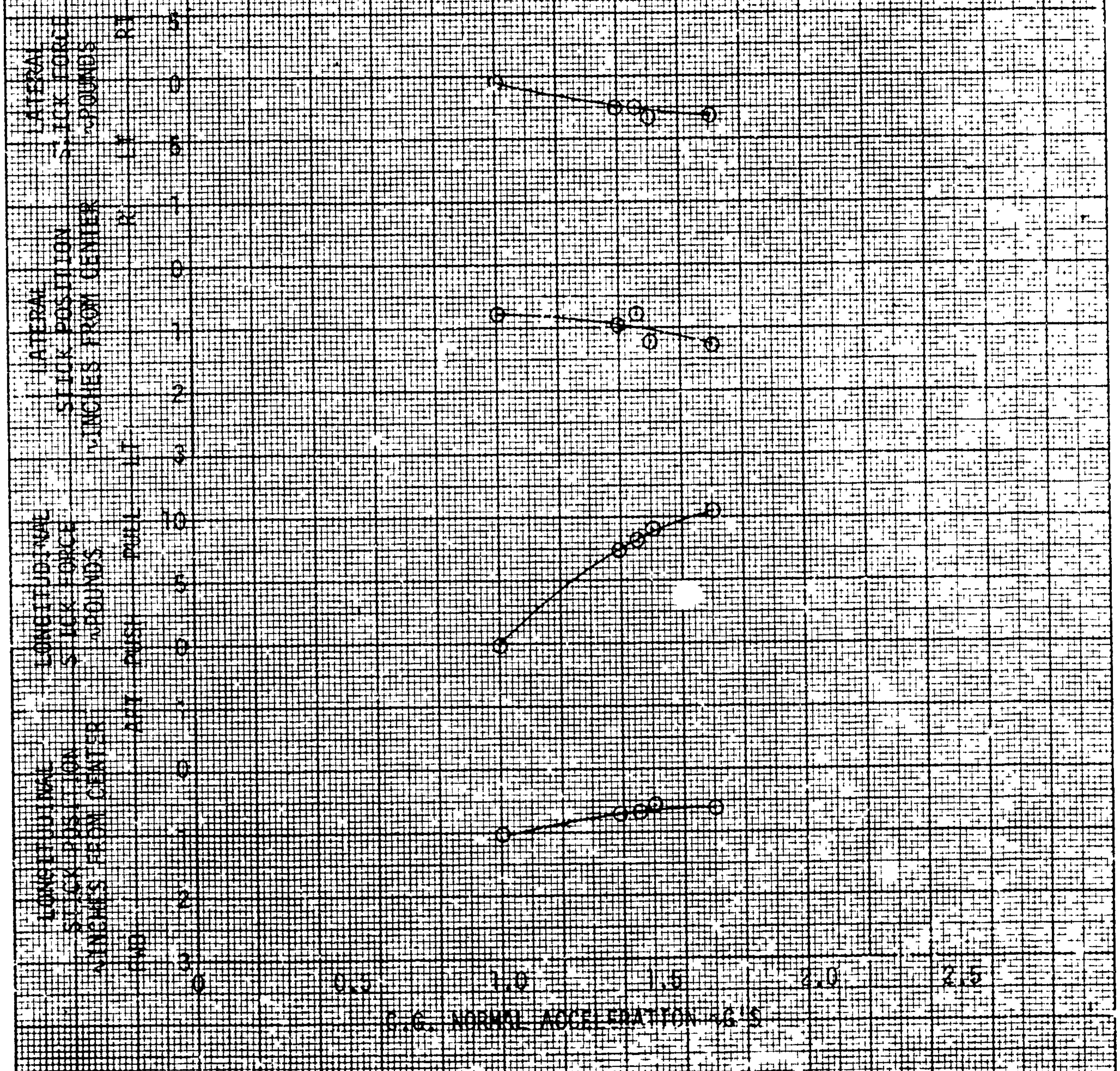


FIGURE 17
MANEUVERING STABILITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
○	20050	299.8	3800	248	5.5	182	EXT STORES GEAR UP	LEFT WIND UP TURNS

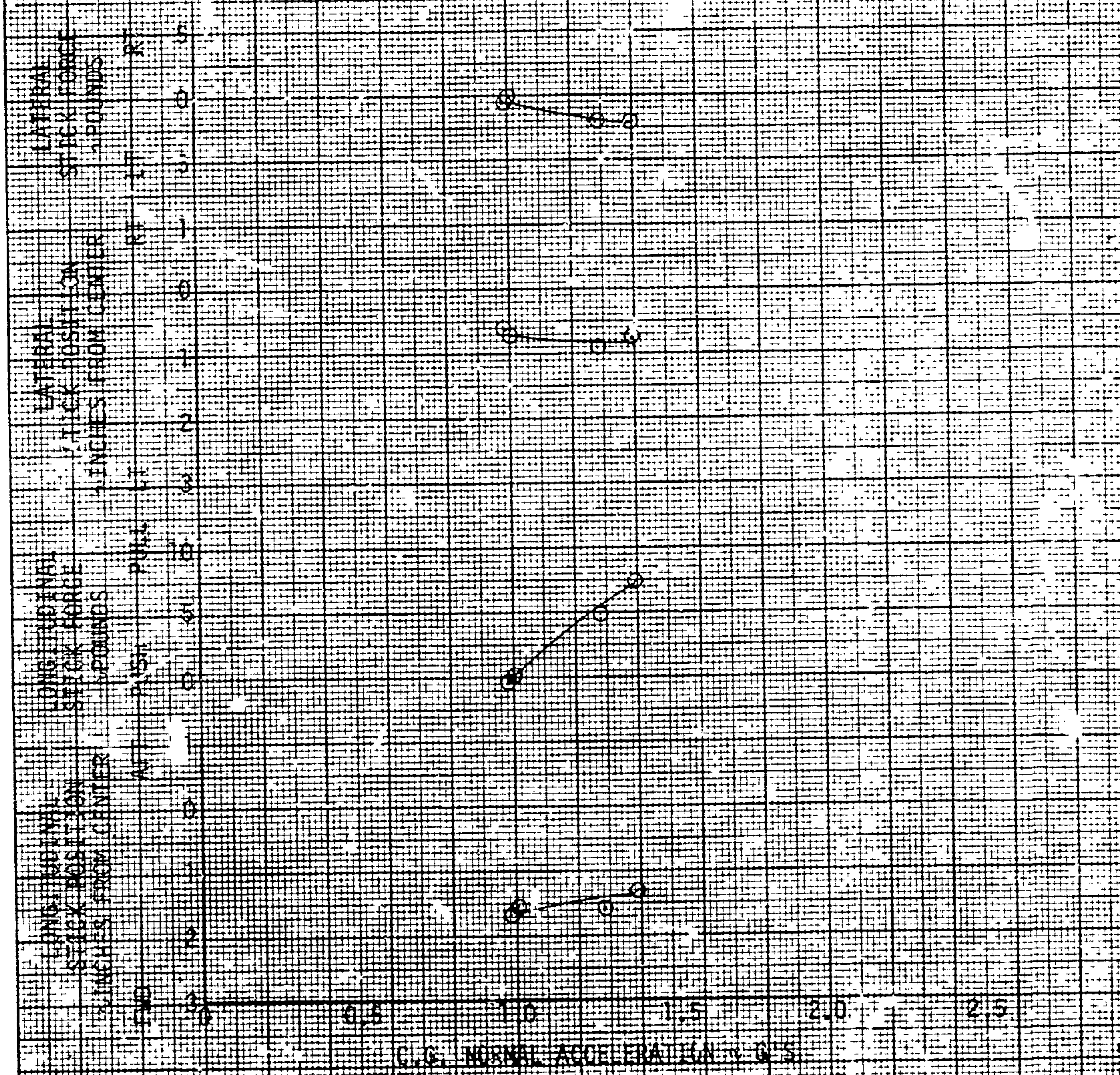


FIGURE 18
BLADE MOMENT STALL
AH-56A USA S/N 66-8834

LONGITUDINAL, PITCH & C.G. ACCEL ————
LATERAL & ROLL - - - - - - -
PEDAL, YAW & SIDESLIP — - —

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~ RPM	COLLECTIVE ANGLE ~DEG	AIRSPEED ~KCAS	CONFIGURATION	MANEUVER
18270	300.1	4760	247	5.8	151	CLEAN GEAR UP	LEFT WIND-UP TURN

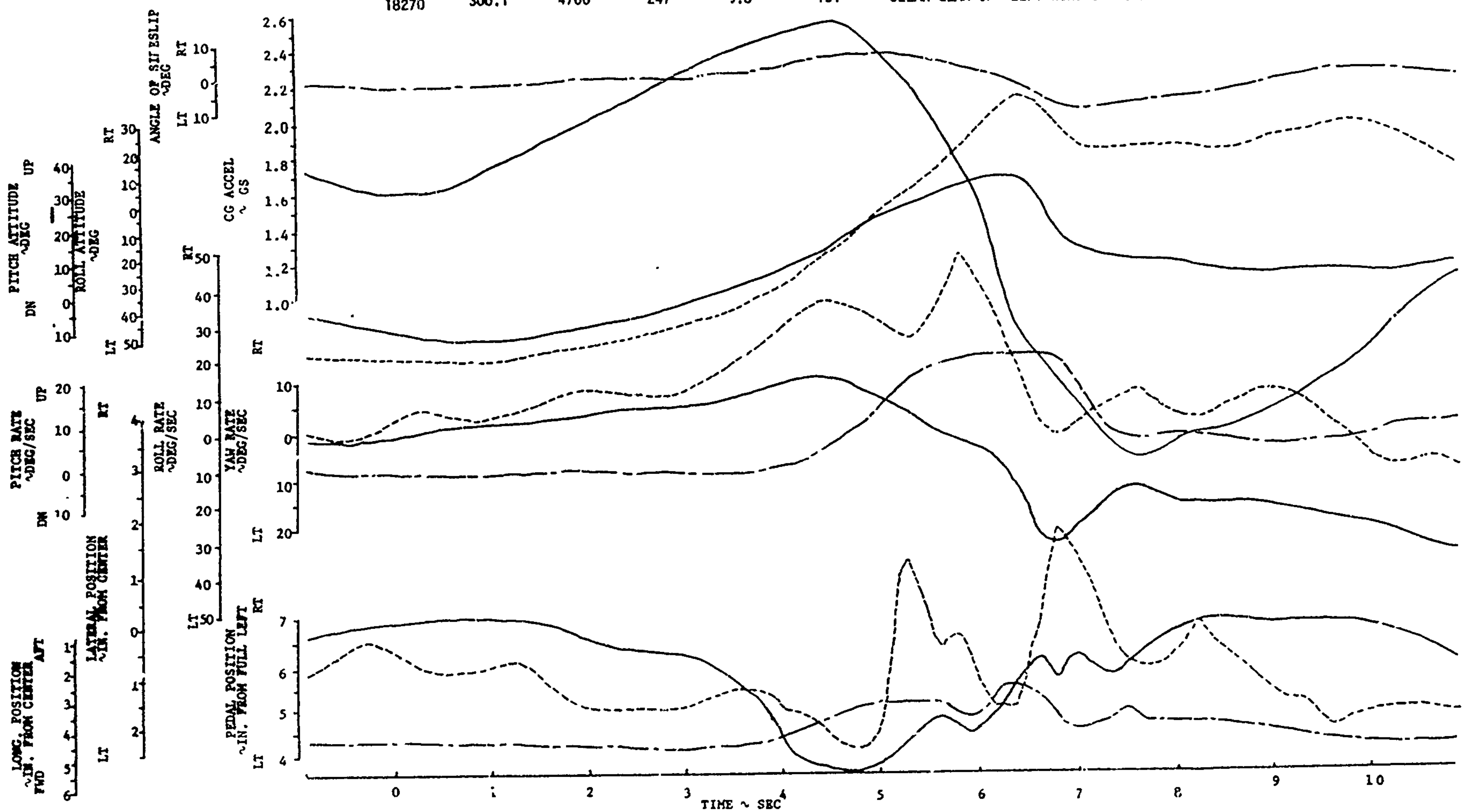
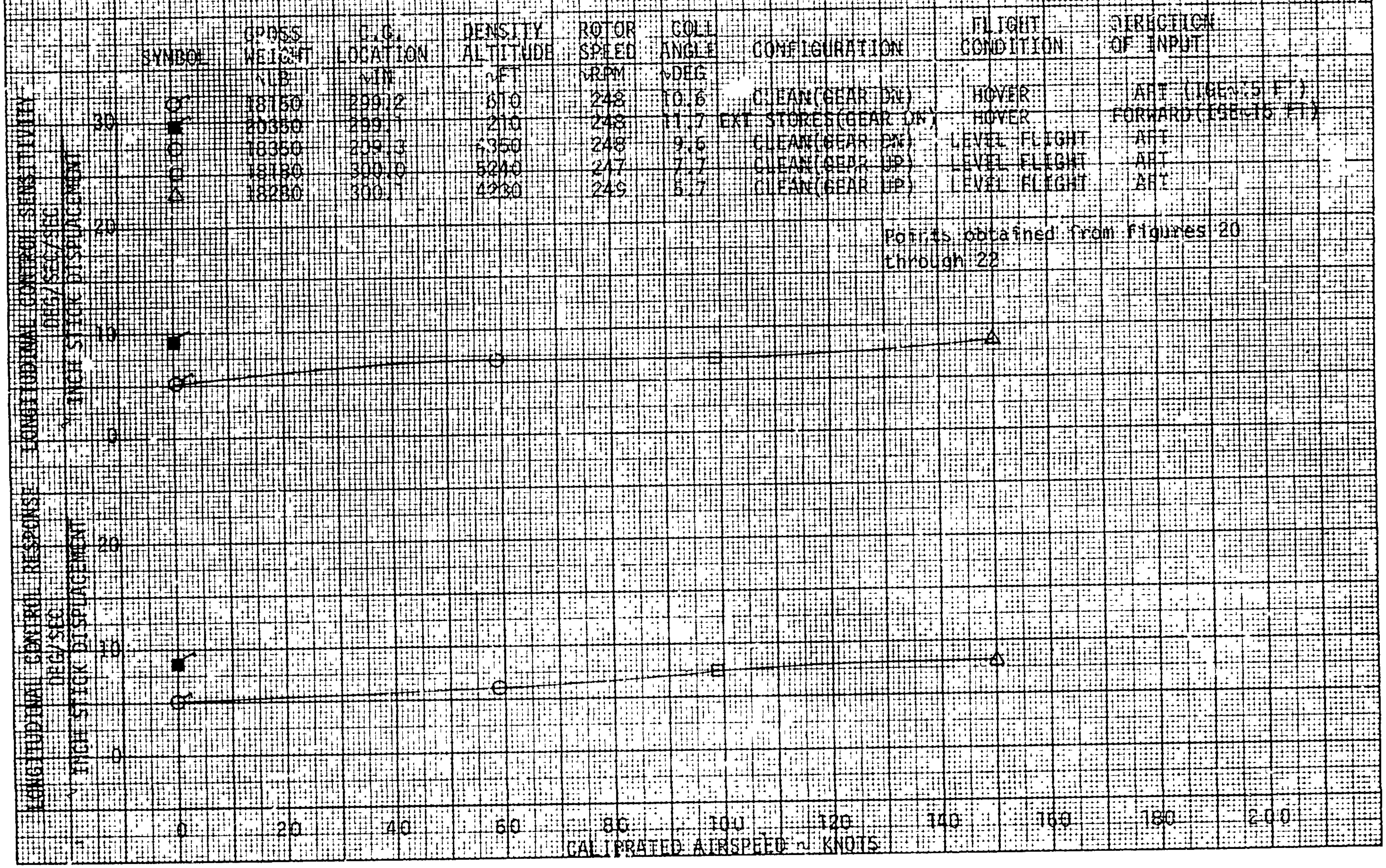


FIGURE 19

SUMMARY LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY

AH-56A USA S/N 66-8834



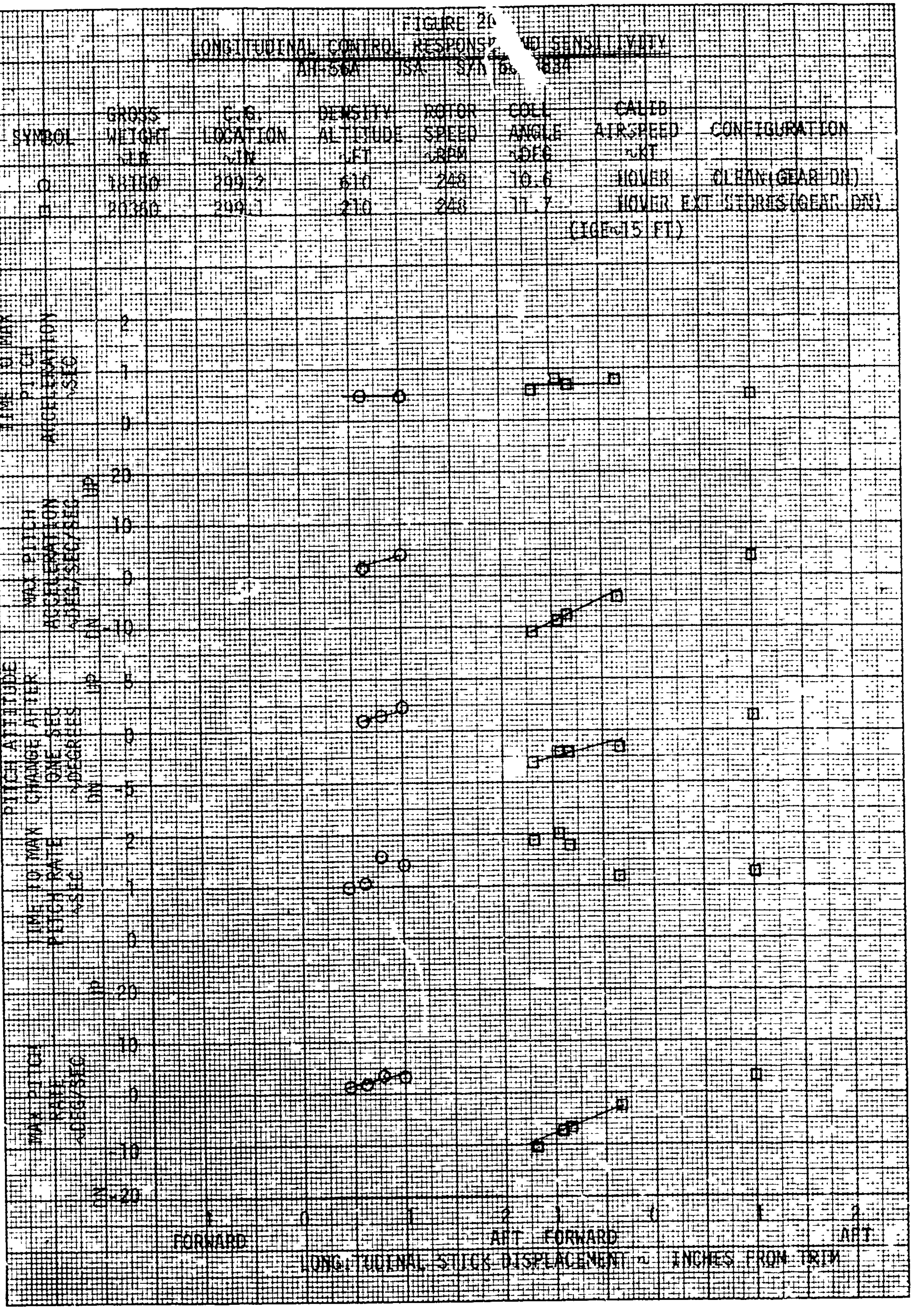


FIGURE 20
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N
SYMBOL
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL ANGLE ~DEG
CALIB AIRSPEED ~KT
CONFIGURATION
○ 18100 299.2 610 248 10.6 HOVER CLEAN(GEAR DN)
□ 20360 299.1 210 248 11.7 HOVER EXT STORES(GEAR DN)
(IGE~15 FT)
TIME TO MAX PITCH ACCELERATION ~SEC
MAX PITCH ACCELERATION ~DEG/SEC/SEC
PITCH ATTITUDE CHANGE AFTER ONE SEC ~DEGREES
TIME TO MAX PITCH RATE ~SEC
MAX PITCH RATE ~DEG/SEC
UP
DN
FORWARD
AFT
LONGITUDINAL STICK DISPLACEMENT ~ INCHES FROM TRIM

FIGURE 21
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
○	18380	299.3	4850	248	9.6	59	CLEAN(GEAR DN)
□	18780	300.0	5240	247	7.7	99	CLEAN(GEAR UP)

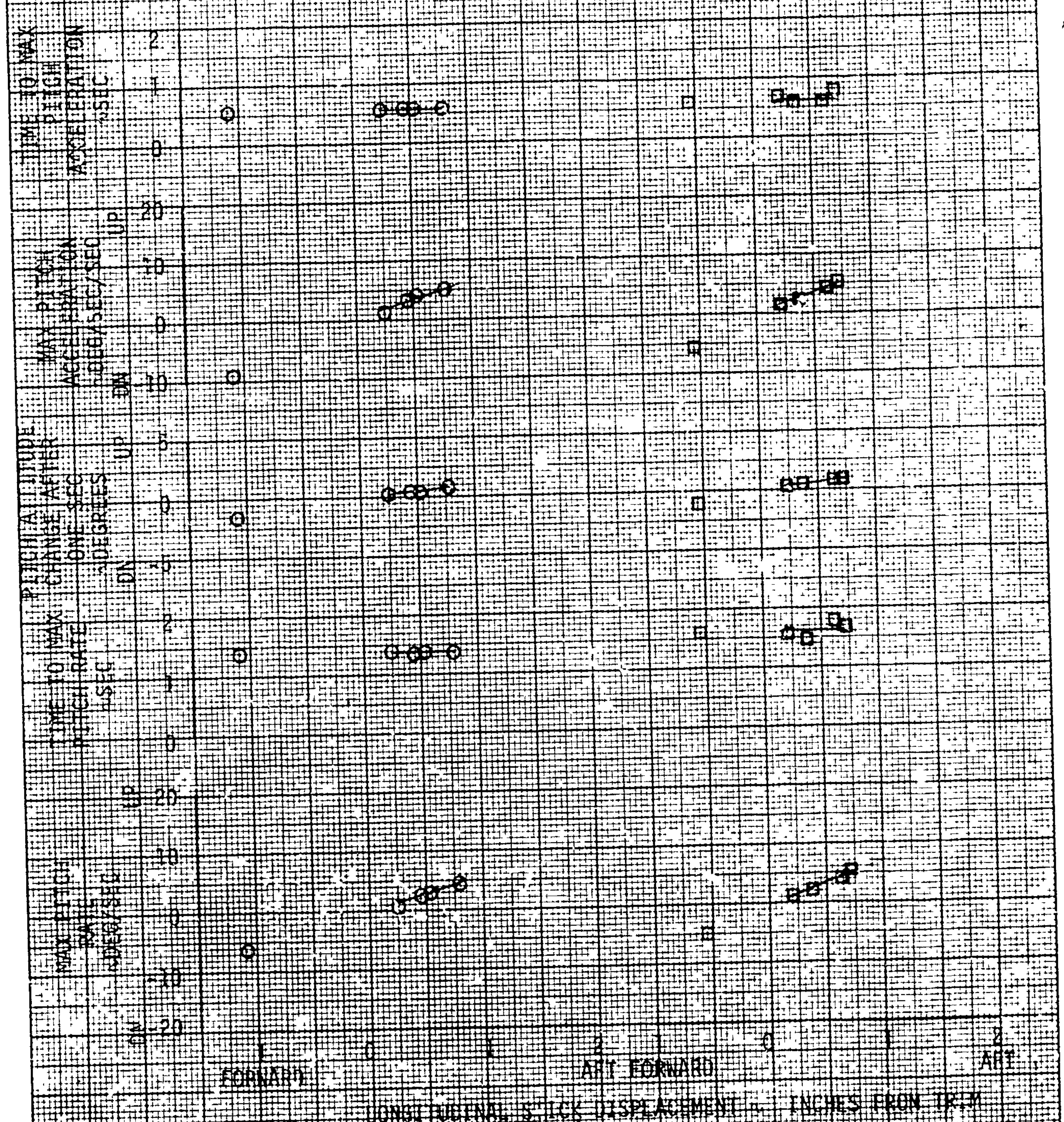


FIGURE 22
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
O	18280	300.1	4230	249	8.7	130	CLEAN(GEAR ON)

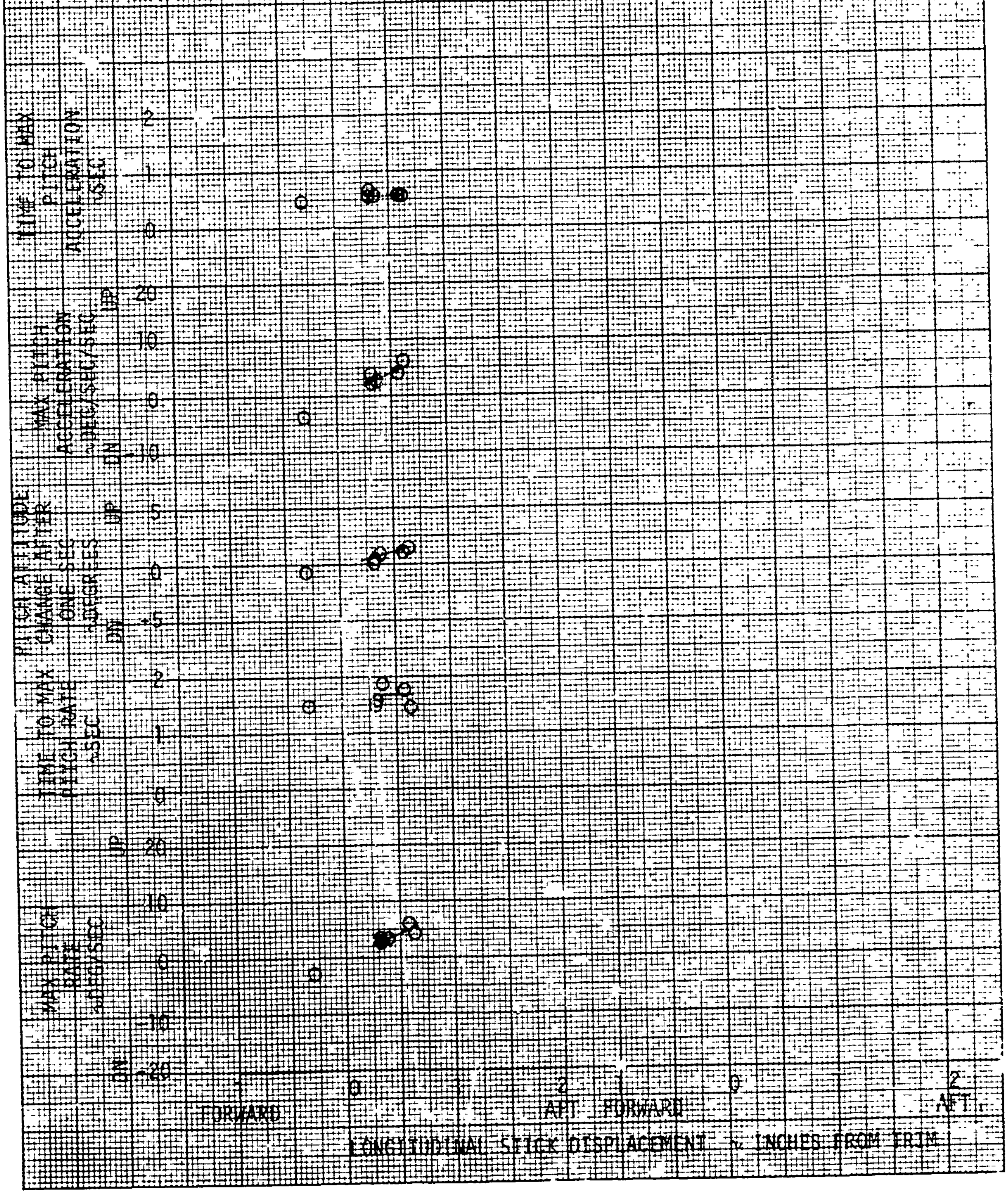


FIGURE 23
SUMMARY LATERAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CONFIGURATION	FLIGHT CONDITION
○ (flagged)	18340	299.3	540	248	10.2	CLEAN(GEAR DN)	HOVER ([illegible] FT)
□ (flagged)	20880	299.2	740	248	11.8	EXT STORES(GEAR DN)	HOVER ([illegible] FT)
○	18500	300.2	4900	250	5.8	CLEAN(GEAR UP)	LEVEL FLIGHT
□	19330	299.9	4840	246	6.7	EXT STORES(GEAR UP)	LEVEL FLIGHT
△	18480	299.3	4700	248	9.7	CLEAN(GEAR DN)	LEVEL FLIGHT
◇	18370	300.1	5130	248	7.6	CLEAN(GEAR UP)	LEVEL FLIGHT

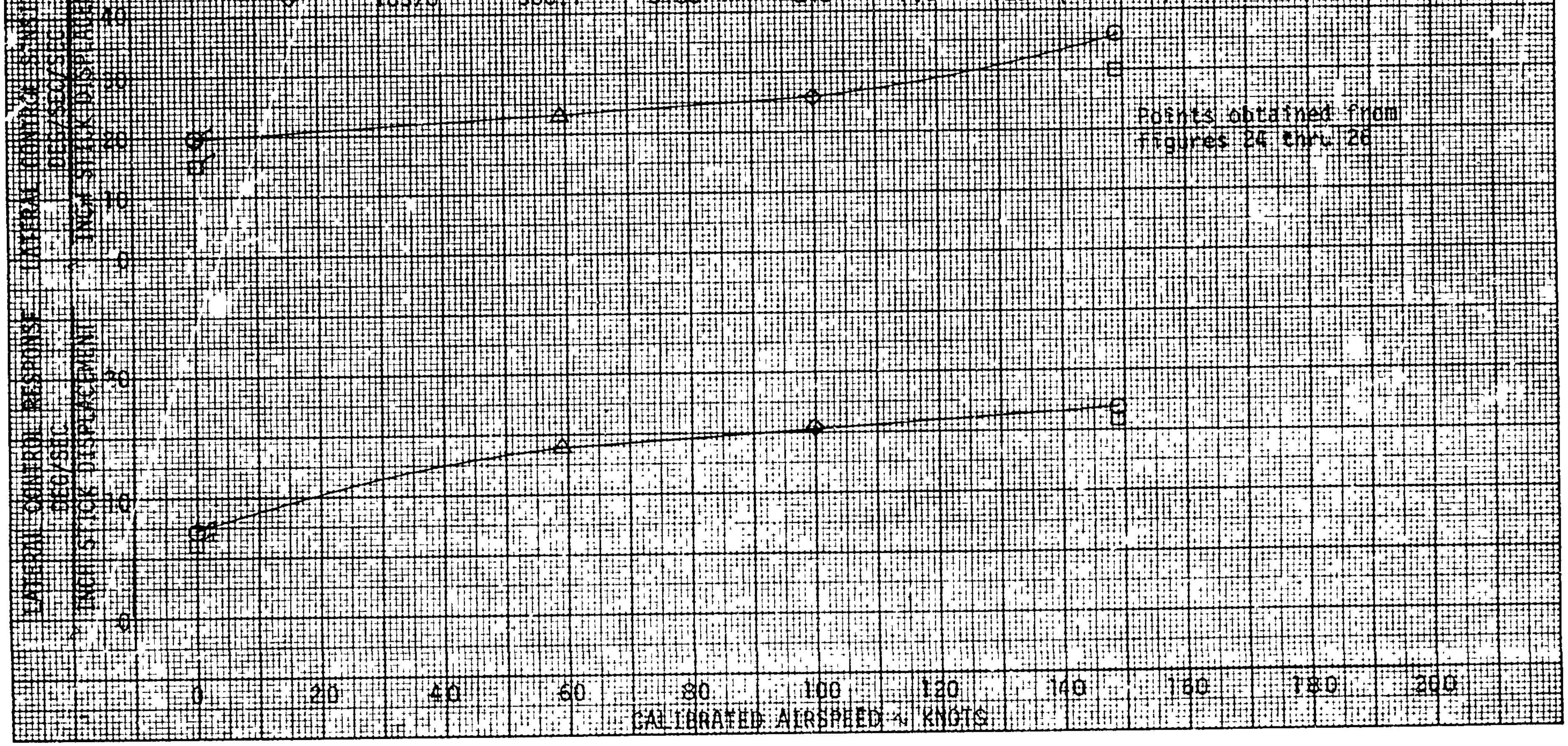


FIGURE 24

LATERAL CONTROL RESPONSE AND SENSITIVITY

AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
○	18340	299.3	540	248	10.2	HOVER	CLEAN(GEAR DN)
□	20600	299.2	140	248	11.8	HOVER	EXT STORES(GEAR DN)
						(IGE~15 FT)	

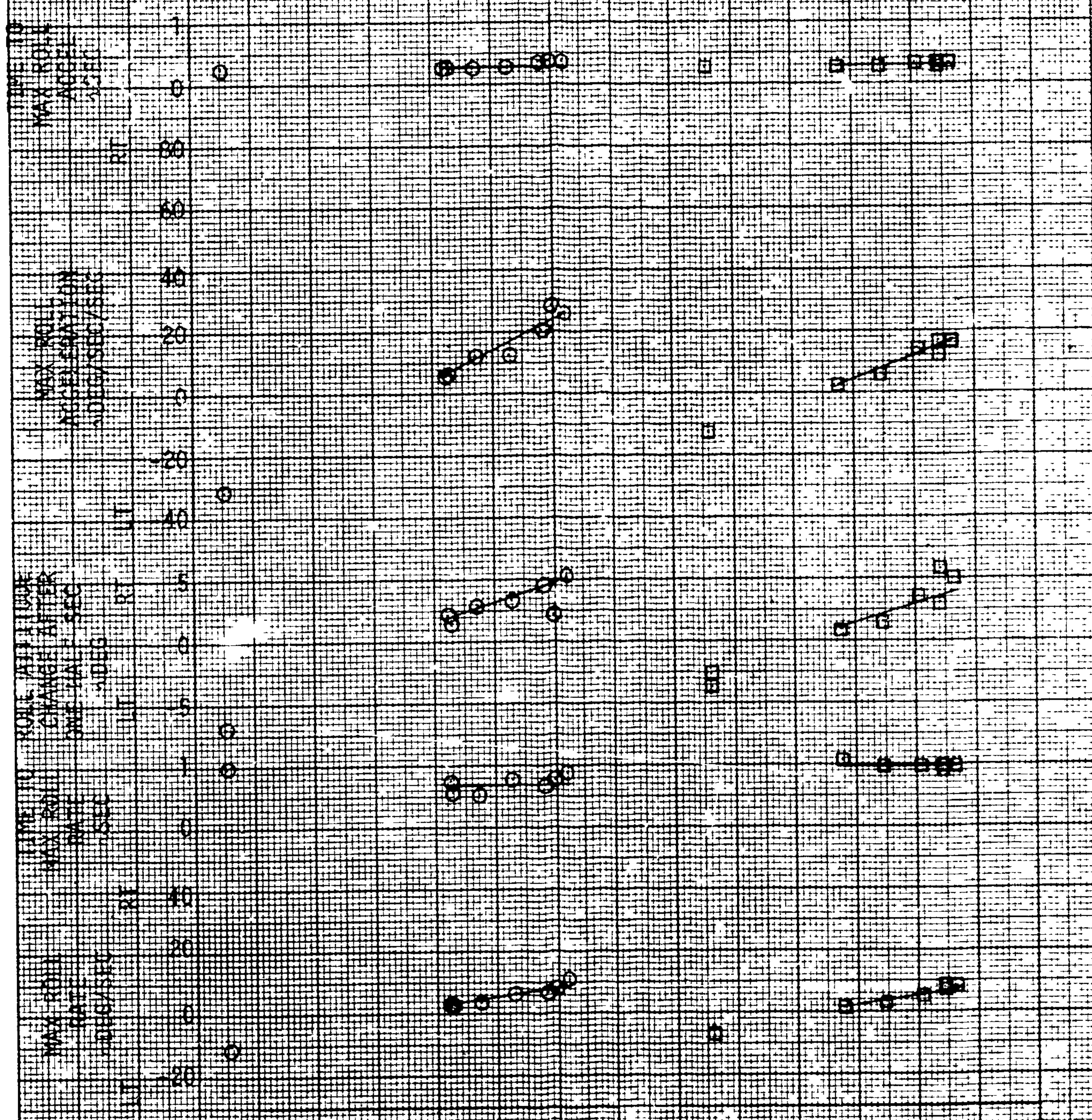


FIGURE 26

LATERAL CONTROL RESPONSE AND SENSITIVITY

AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
○	18480	299.6	4700	248	9.7	59	CLEAN(GEAR DN)
□	18370	[illegible]	5130	247	7.6	100	CLEAN(GEAR UP)

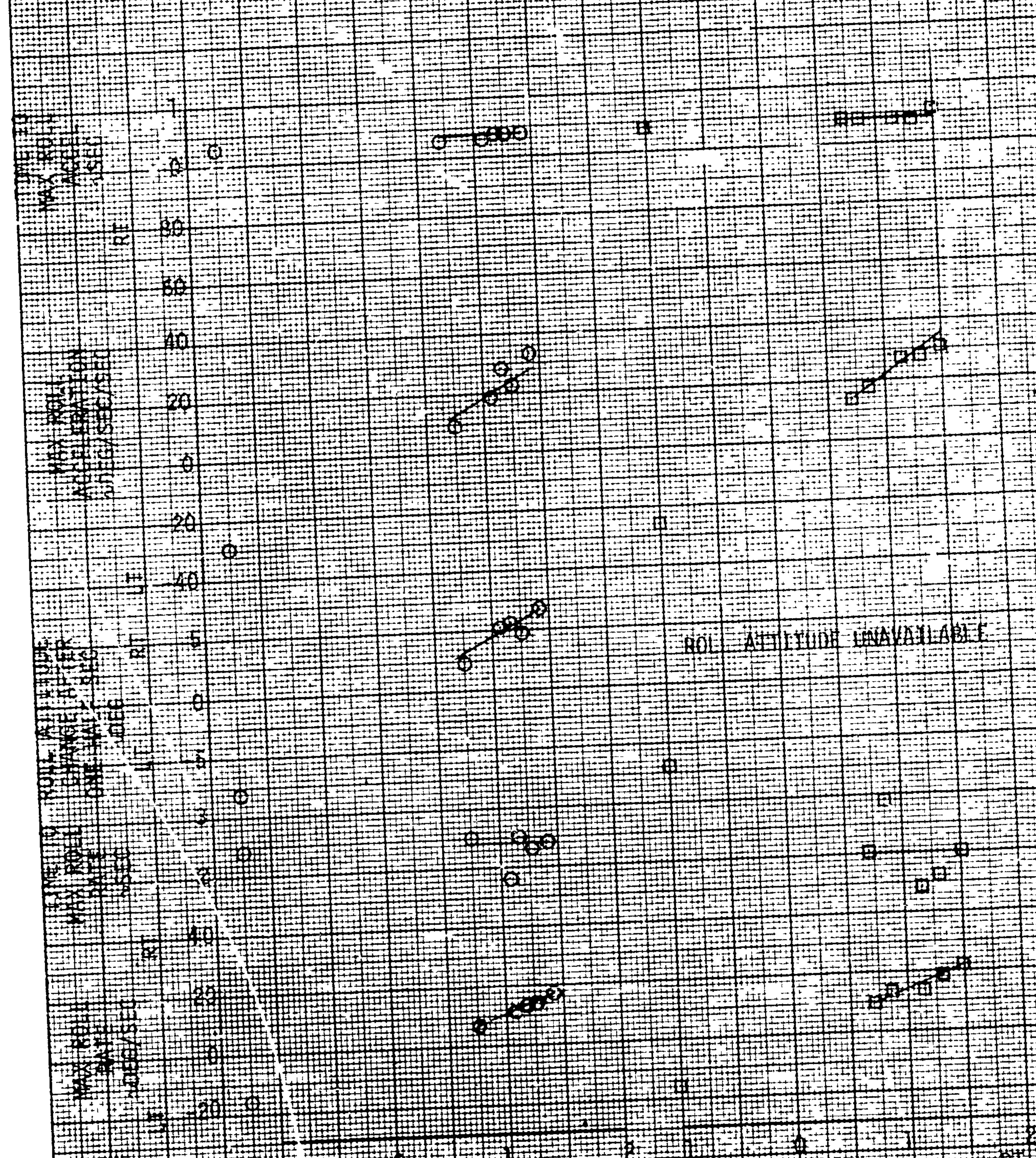
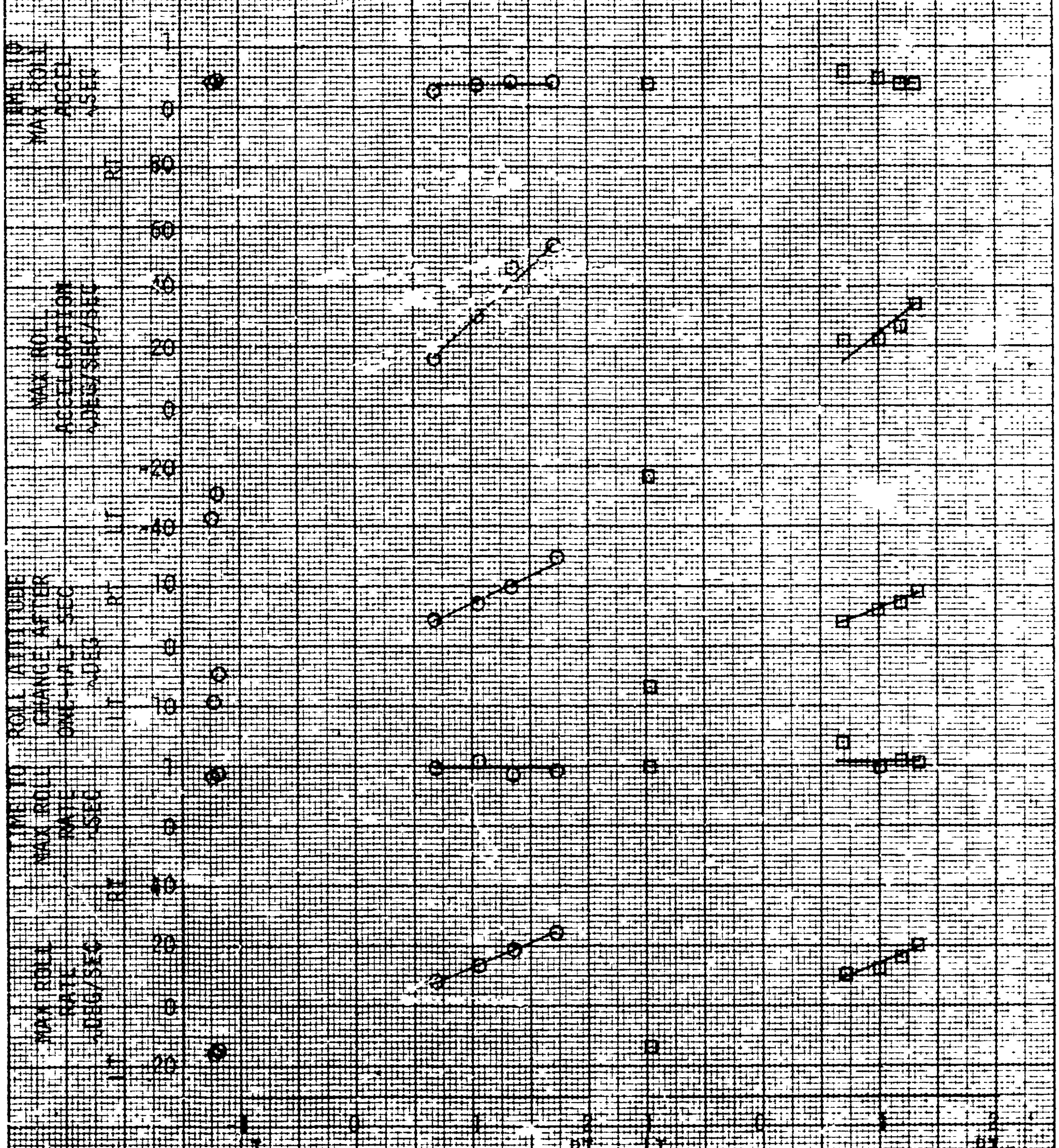


FIGURE 26
LATERAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
○	18500	300.2	4900	250	5.8	149	CLEAN(GEAR UP)
□	19380	299.9	4840	246	5.7	149	EXT STORES(GEAR UP)



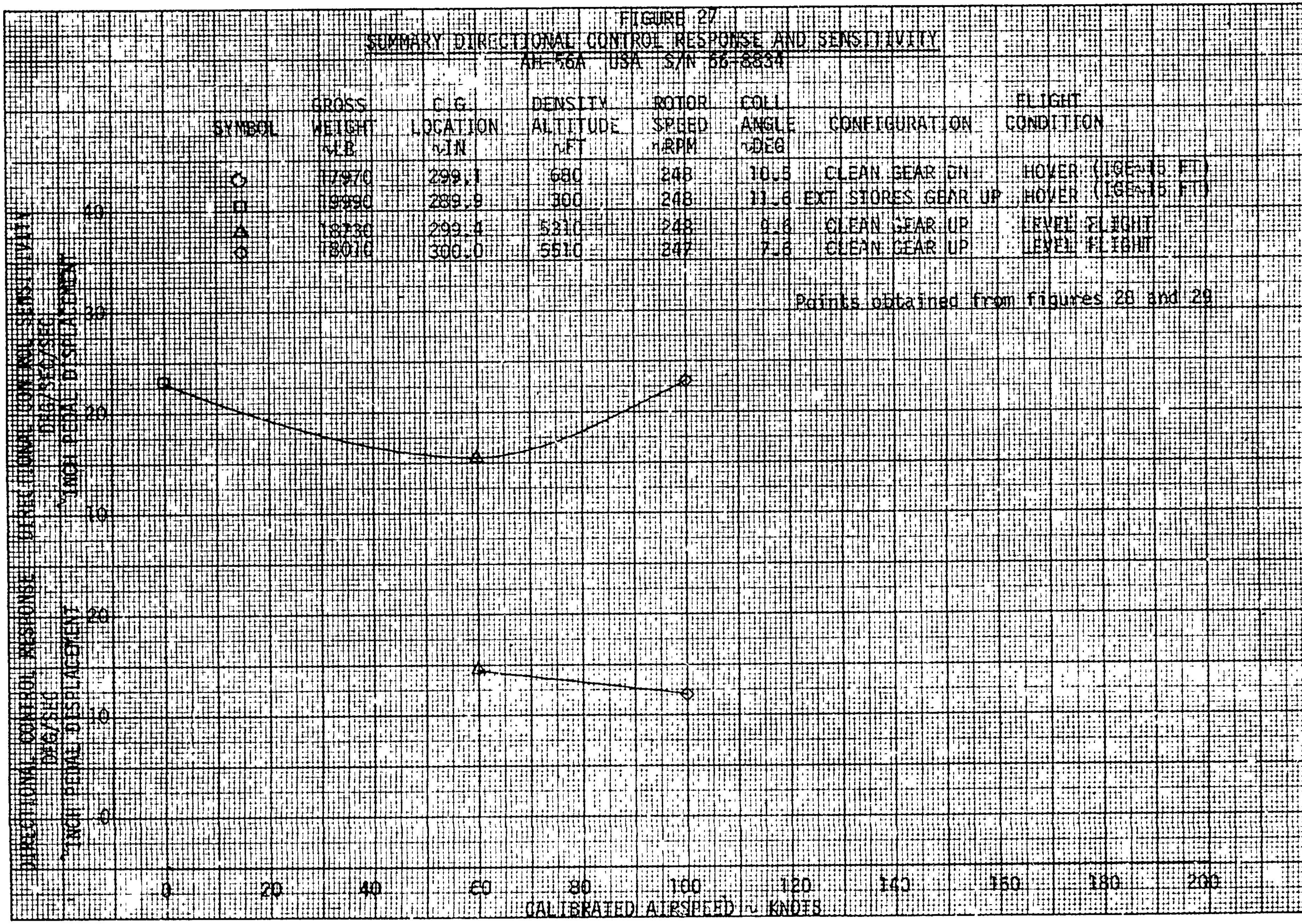


FIGURE 27
SUMMARY DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834
SYMBOL
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL ANGLE ~DEG
CONFIGURATION
FLIGHT CONDITION
17970 299.1 680 248 10.5 CLEAN GEAR ON HOVER (IGE~10 FT)
19990 289.9 300 248 11.6 EXT STORES GEAR UP HOVER (IGE~15 FT)
18230 299.4 5310 248 8.6 CLEAN GEAR UP LEVEL FLIGHT
18010 300.0 5510 247 7.8 CLEAN GEAR UP LEVEL FLIGHT
Points obtained from figures 28 and 29
DIRECTIONAL CONTROL SENSITIVITY
DEG/SEC/SEC
~INCH PEDAL DISPLACEMENT
DIRECTIONAL CONTROL RESPONSE
DEG/SEC
~INCH PEDAL DISPLACEMENT
40
30
20
10
20
10
0
0
20
40
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

A.P.E. 1.3B

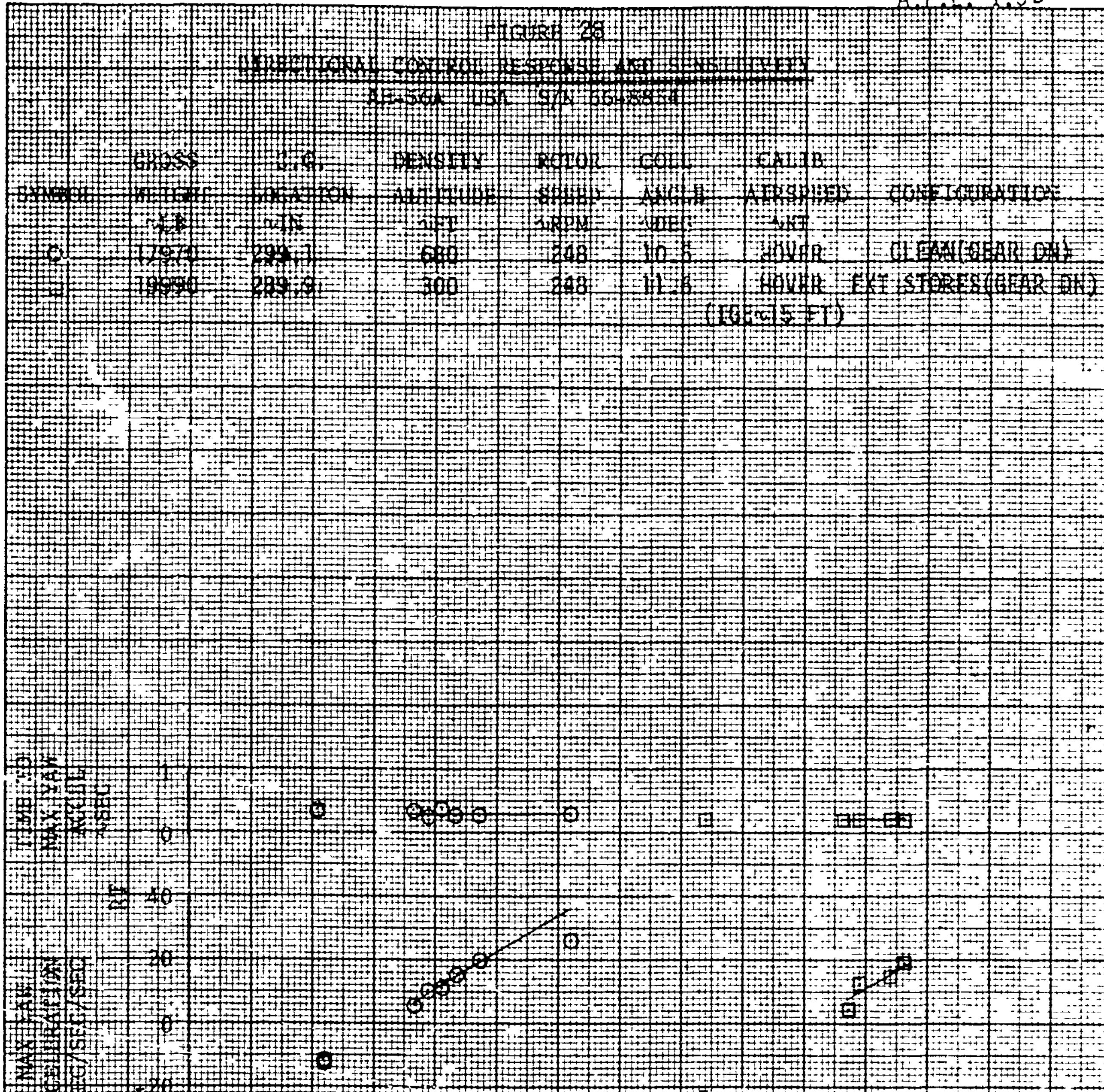


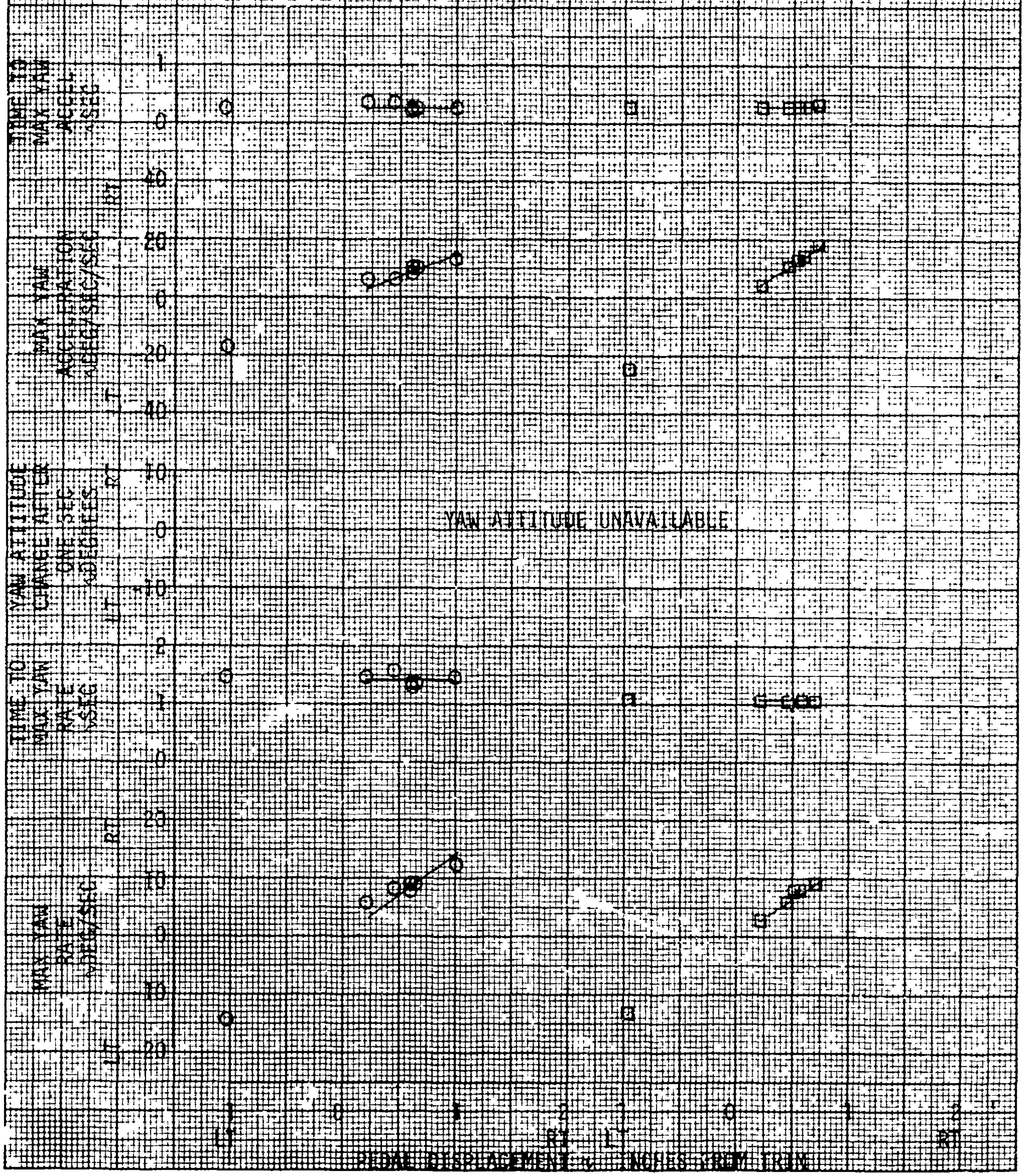
FIGURE 28
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834
SYMBOL
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
COLL ANGLE ~DEG
CALIB AIRSPEED ~KT
CONFIGURATION
17970
299.1
680
248
10.6
HOVER
CLEAN(GEAR ON)
19990
289.9
300
248
11.6
HOVER
EXT STORES(GEAR ON)
(IGE~15 FT)
MAX YAW ACCELERATION ~DEG/SEC/SEC
YAW RATE AT ONE SEC ~DEG/SEC
40
20
0
-20
-40
20
10
0
-10
-20
PEDAL DISPLACEMENT ~ INCHES FROM TRIM
1
0
1
2
LT
RT

FIGURE 29

DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY

AH-56A USA S/N 66-8834

SYMBOL	GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
○	18730	299.4	5310	248	9.6	60	CLEAN(GEAR DN)
□	18010	300.0	5510	247	7.6	100	CLEAN(GEAR UP)

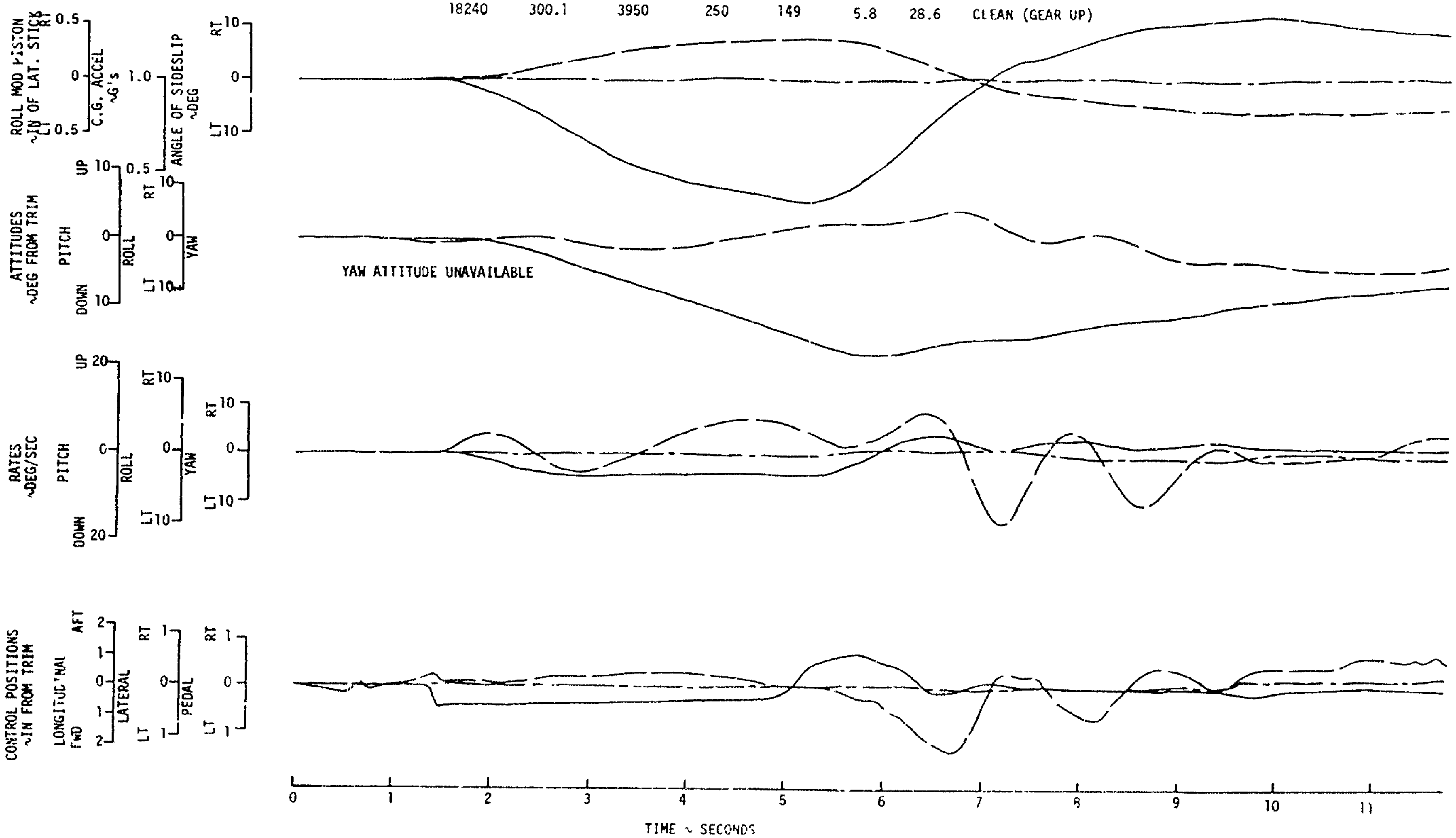


A.P.E. I.3B

FIGURE 30
FORWARD LONGITUDINAL STEP
AH-56A USA S/N 66-8834

LONGITUDINAL, PITCH & C.G. ACCEL ————
ROLL MOD PISTON, LATERAL & ROLL — — —
PEDAL, YAW & SIDESLIP —·—·—

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
18240	300.1	3950	250	149	5.8	28.6	CLEAN (GEAR UP)

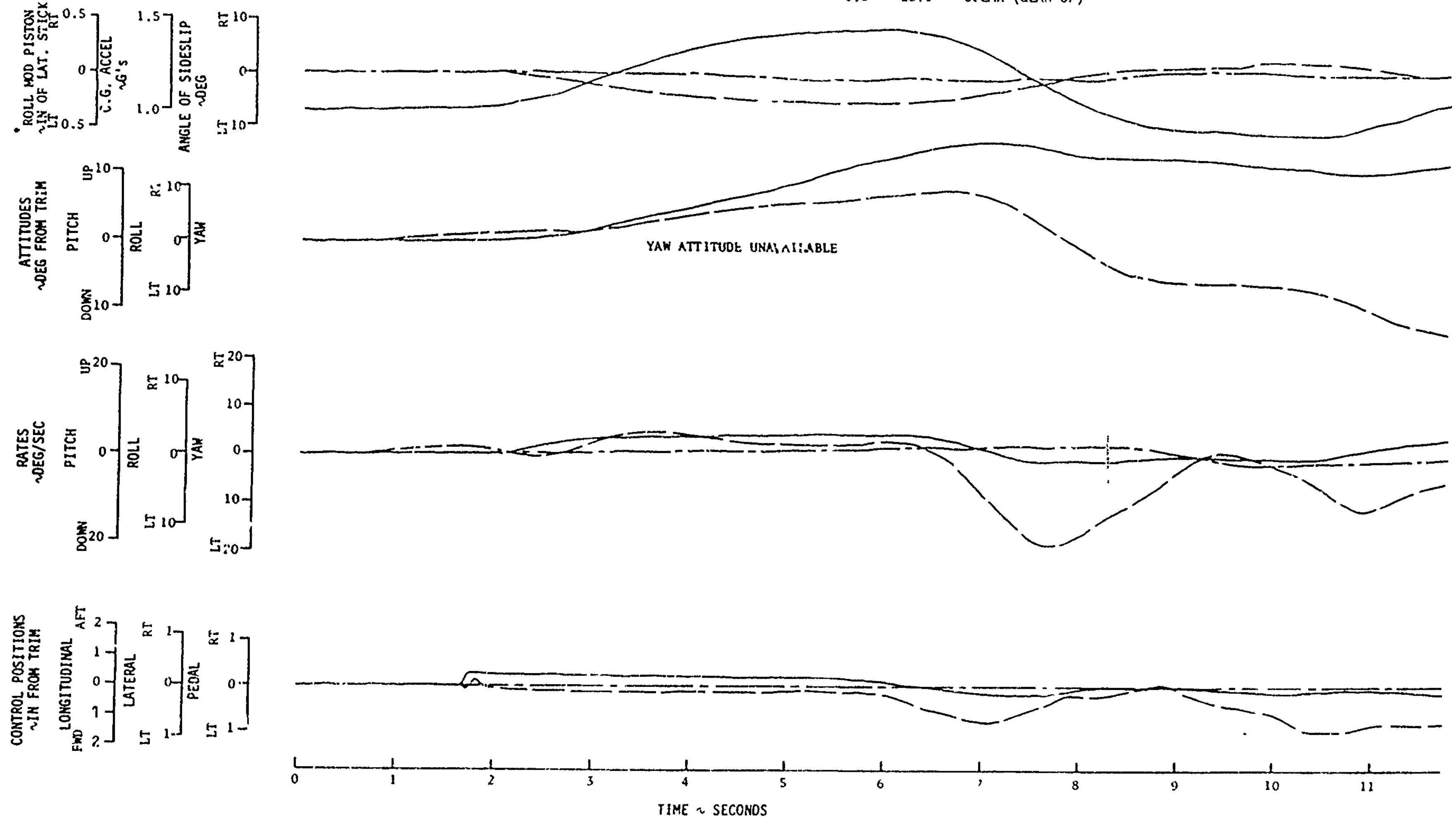


A.P.E I 3B

FIGURE 3.
AFT LONGITUDINAL STEP
AH-56A USA S/N 66-8834

LONGITUDINAL, PITCH & C.G. ACCEL ——·——
ROLL MOD PISTON, LATERAL & ROLL —— ——
PEDAL, YAW & SIDESLIP ——·· ——

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
18440	299.4	1750	250	150	5.8	28.6	CLEAN (GEAR UP)



A.P.E. 1.38

FIGURE 3[illegible]

RIGHT LATERAL SIDESLIP

AH-56A USA S/N 66-8834

LONGITUDINAL, PITCH & C.G. ACCEL ————

LATERAL & ROLL — — —

PEDAL, YAW & SIDESLIP — - —

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
18540	300.2	5010	250	153	5.8	28.6	CLEAN (GEAR UP)

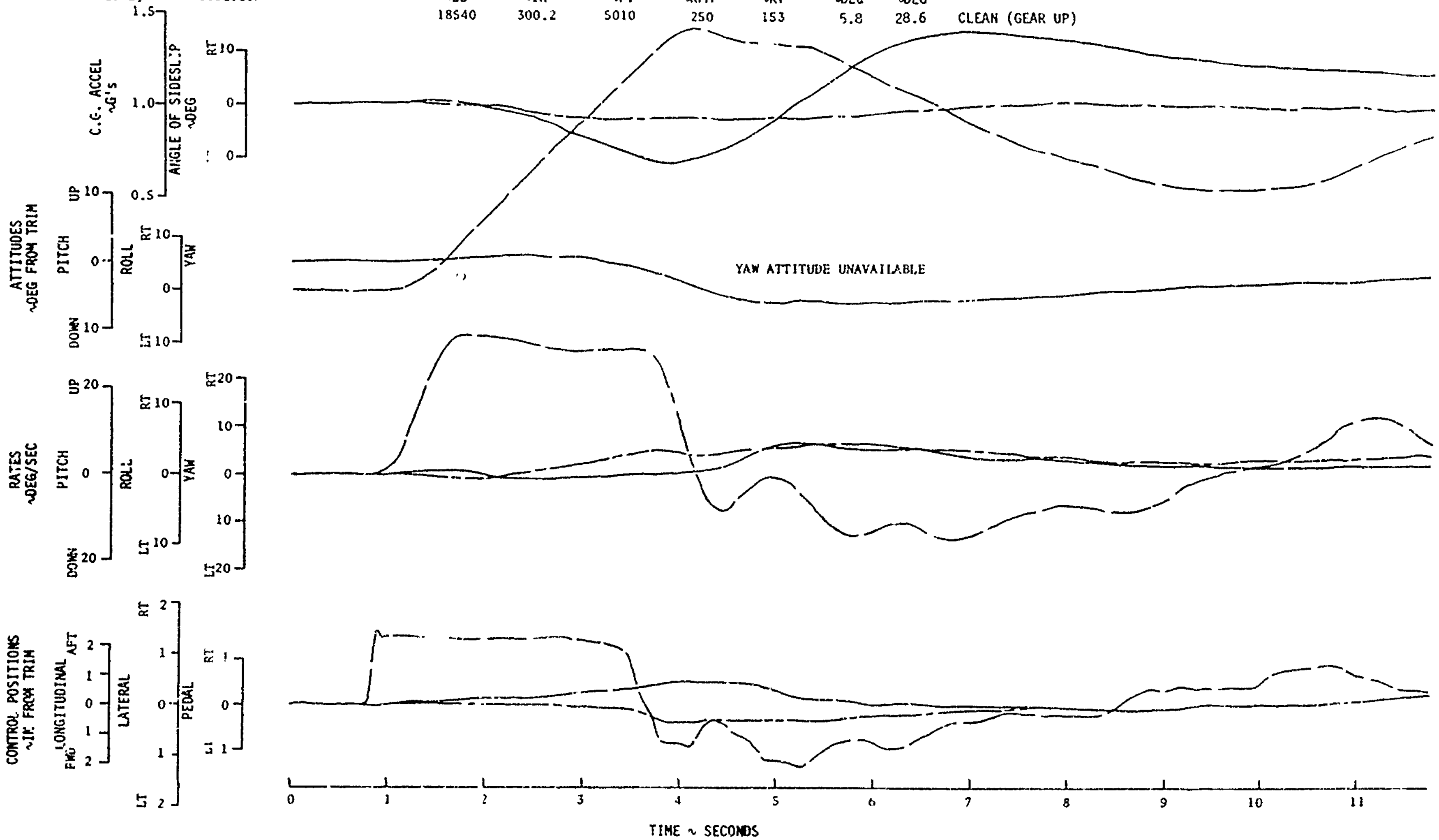
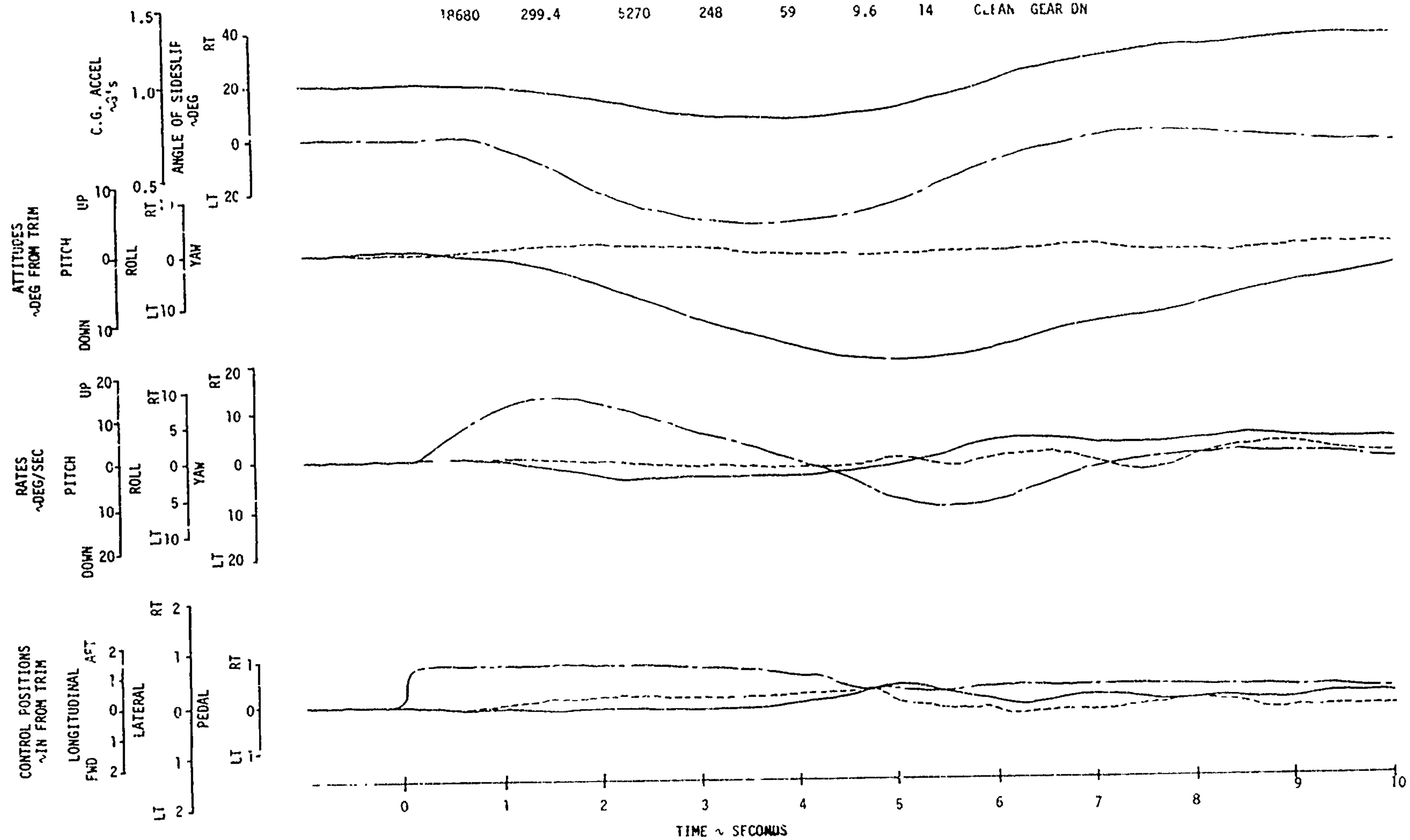


FIGURE 33
RIGHT DIRECTIONAL STEP
AH-56A USA S/N 65-8834

LONGITUDINAL, PITCH & C.G. ACCEL
LATERAL & ROLL
PEDAL, YAW & SIDESLIP

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
18680	299.4	5270	248	59	9.6	14	CLEAN GEAR DN



A.P.E. 1.3B

FIGURE 34
RIGHT LATERAL PULSE
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
18250	299.2	570	249	HOVER	10.2	-2.2	CLEAN (GEAR DN)

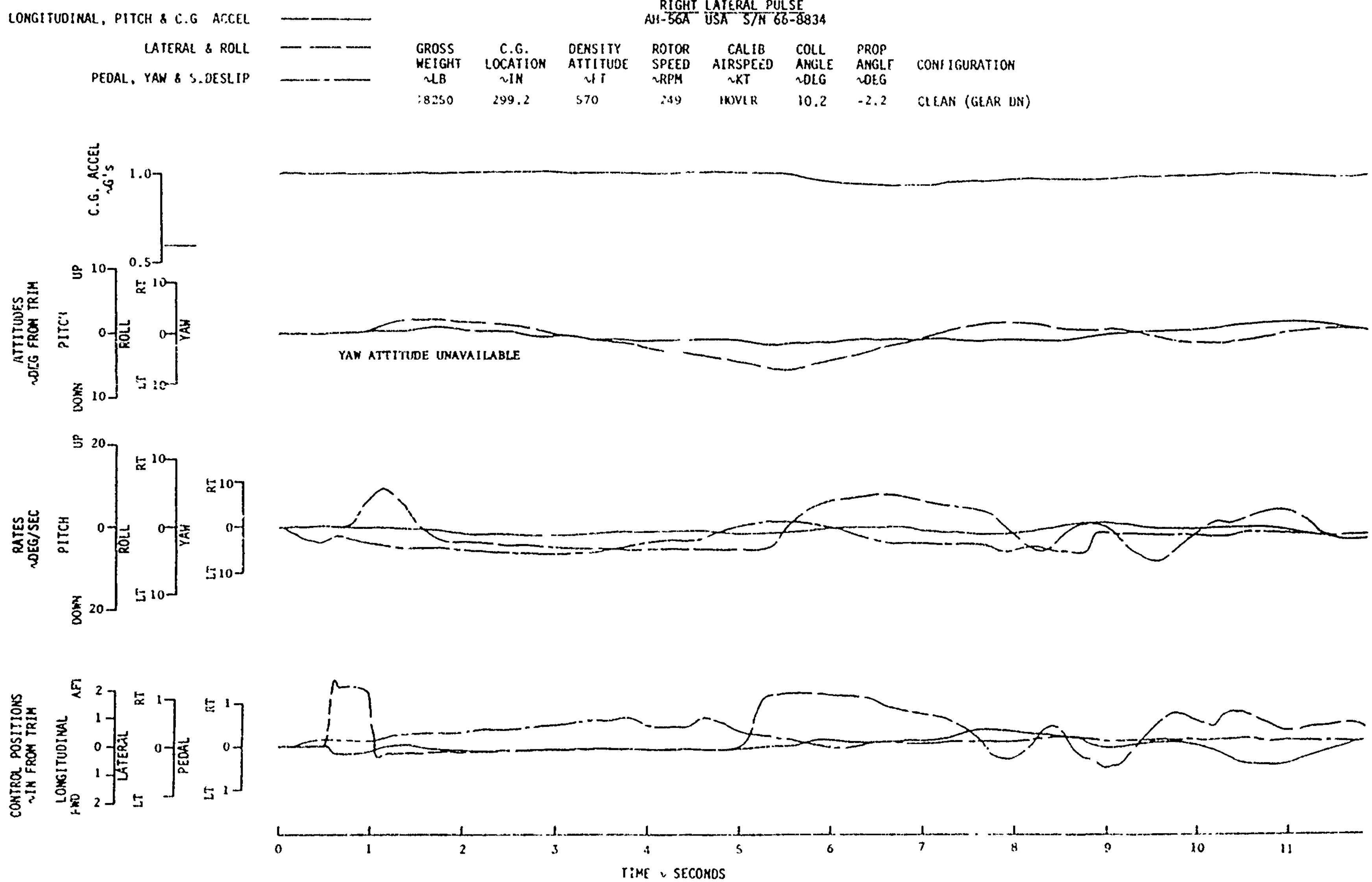
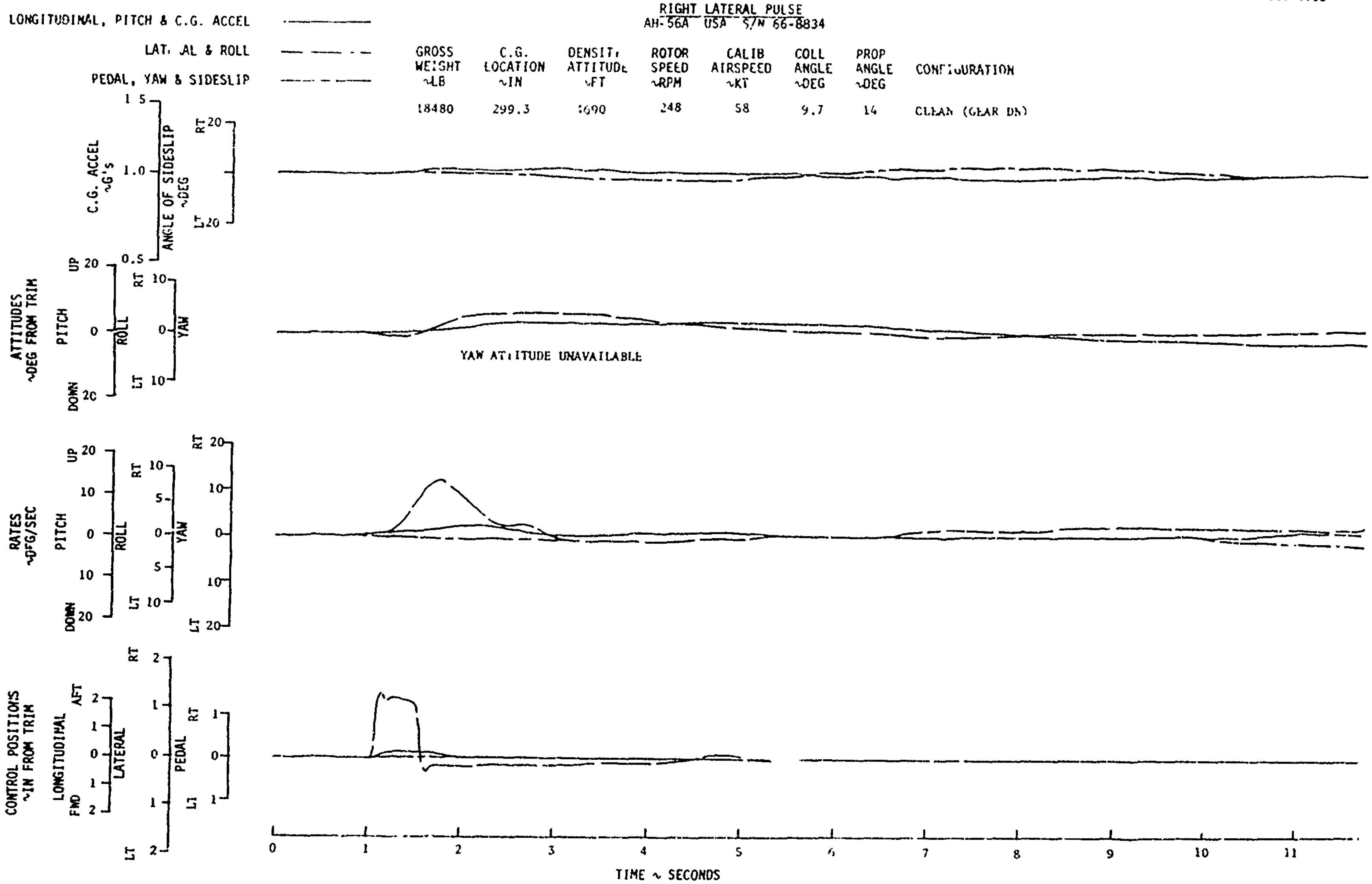


FIGURE 35

RIGHT LATERAL PULSE

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
18480	299.3	[illegible]690	248	58	9.7	14	CLEAN (GEAR DN)



LONGITUDINAL, PITCH & COLLECTIVE
LATERAL, ROLL & PROP
PEDAL & YAW

A.P.E. I 3B

FIGURE 36

LANDING

AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION
18150	299.0	450	247	CLEAN GEAR ON

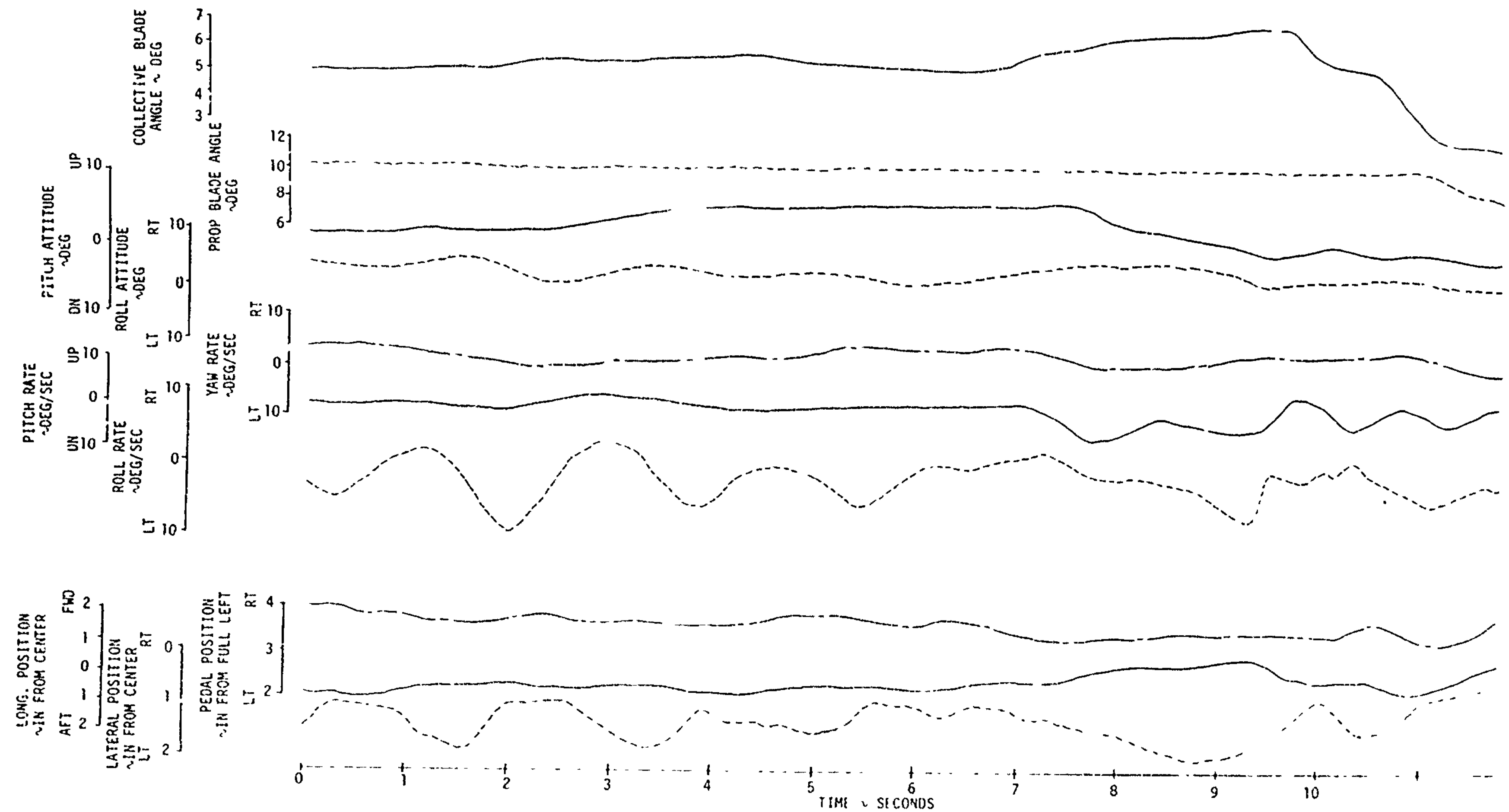


FIGURE 37
CONTROL POSITIONS IN SIDEWARD FLIGHT
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	PROP ANGLE ~DEG	WHEEL HEIGHT ~FT	CONFIGURATION
20840	293(FWD)	750	247	-2.2	5	EXT. STORES GEAR DOWN

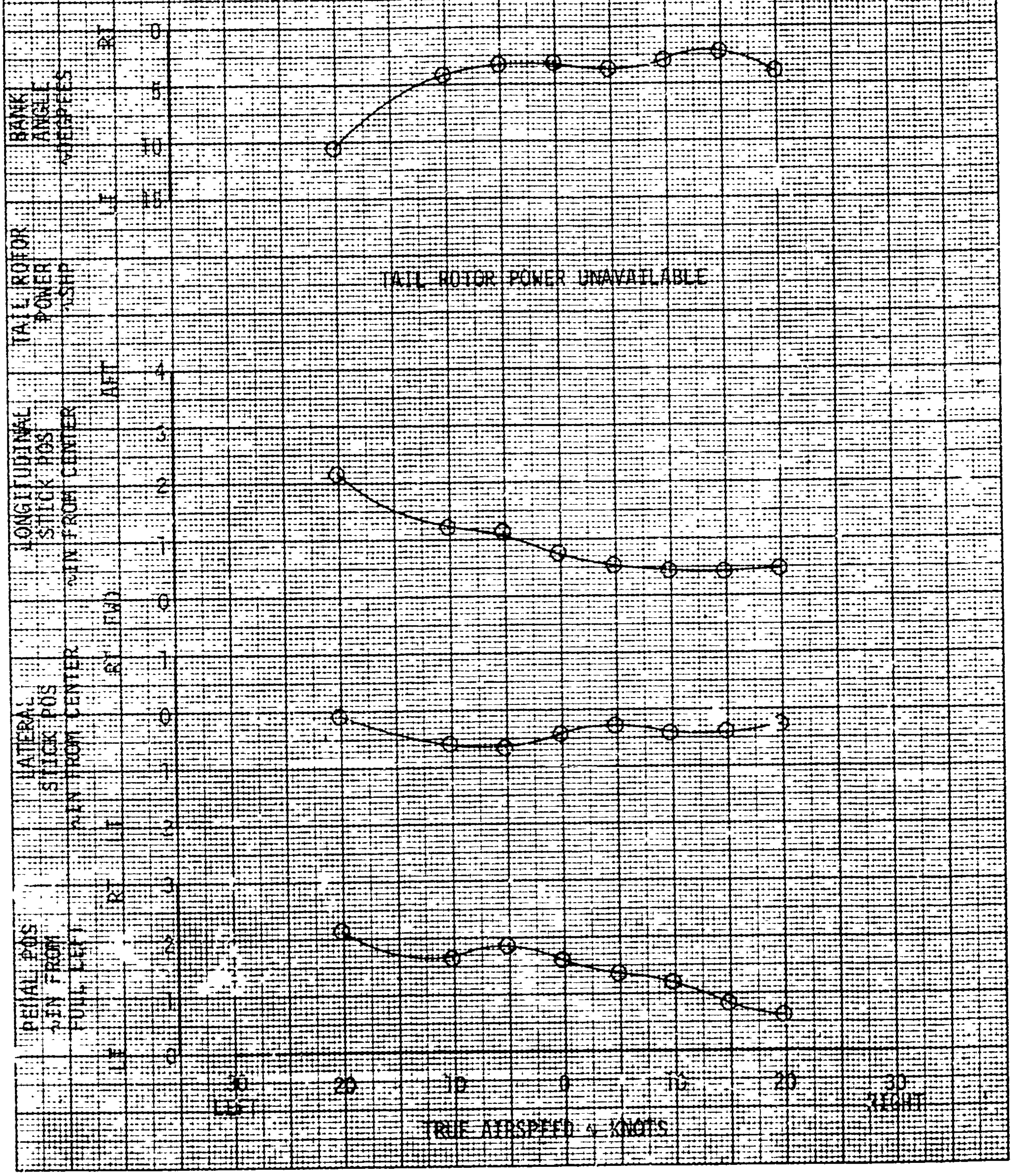
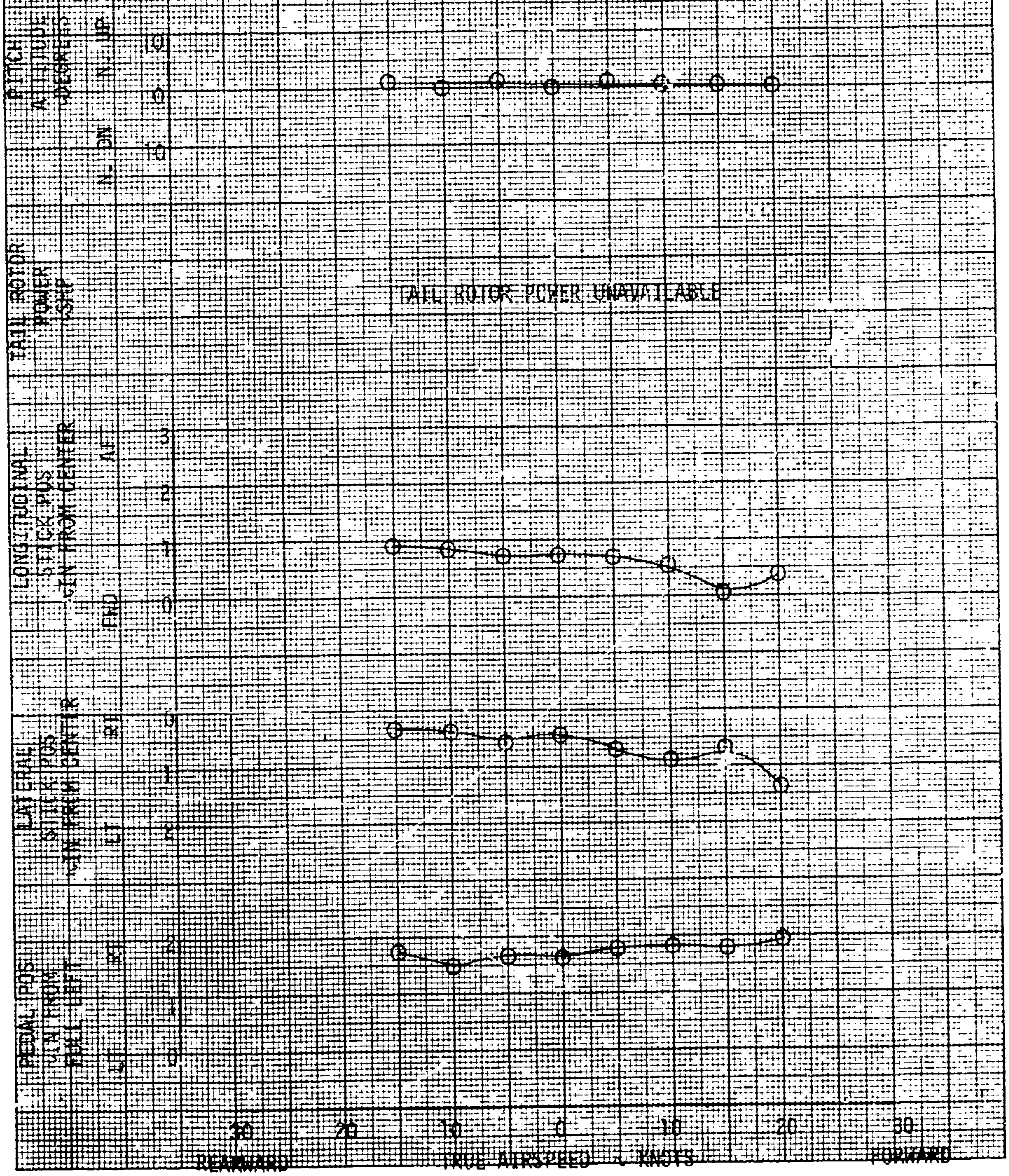


FIGURE 38
CONTROL POSITIONS IN REARWARD FLIGHT
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	PROP ANGLE ~DEG	WHEEL HEIGHT ~FT	CONFIGURATION
20280	293(FWD)	750	248	-2.2	5	EXT. STORES GEAR DOWN



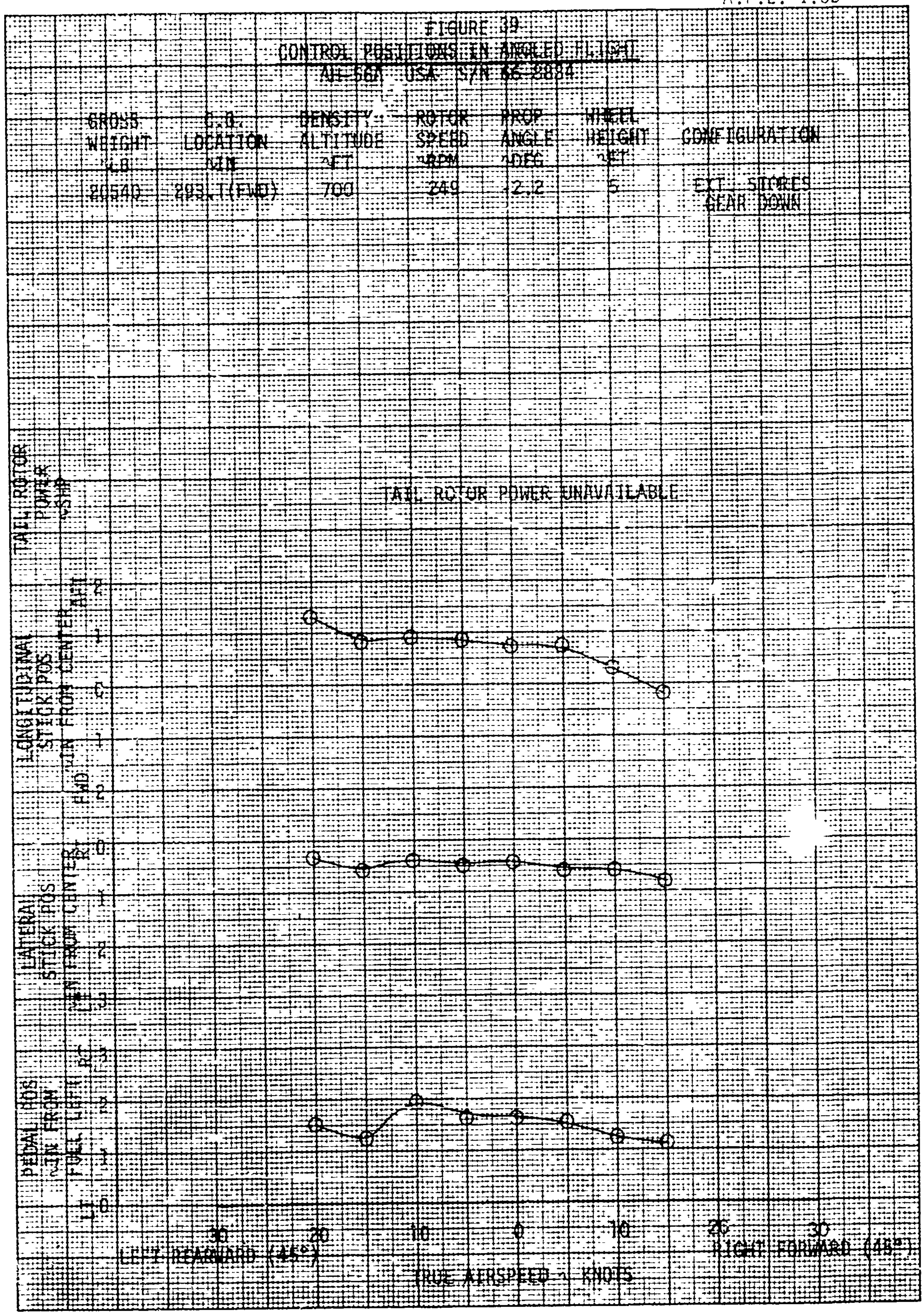


FIGURE 39
CONTROL POSITIONS IN ANGLED FLIGHT
AH-56A USA S/N 66-8834
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
PROP ANGLE ~DEG
WHEEL HEIGHT ~FT
CONFIGURATION
20540
293.1(FWD)
700
249
-2.2
5
EXT. STORES
GEAR DOWN
TAIL ROTOR POWER ~SHP
TAIL ROTOR POWER UNAVAILABLE
LONGITUDINAL STICK POS ~IN FROM CENTER
FWD
AFT
LATERAL STICK POS ~IN FROM CENTER
LT
RT
PEDAL POS ~IN FROM FULL LEFT
LEFT REARWARD (45°)
RIGHT FORWARD (45°)
TRUE AIRSPEED ~ KNOTS

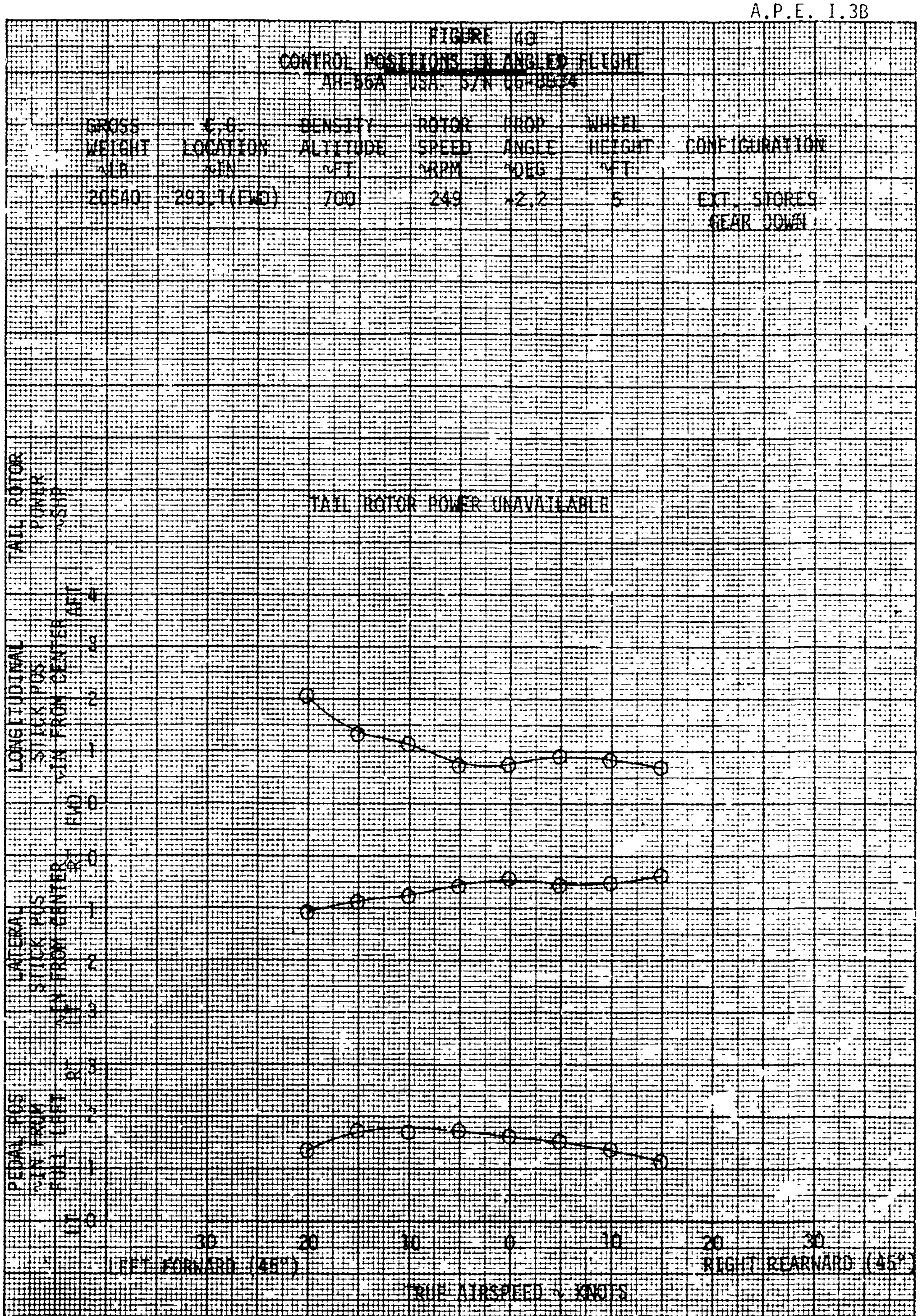


FIGURE
CONTROL POSITIONS IN ANGLED FLIGHT
AH-56A USA S/N
GROSS WEIGHT ~LB
C.G. LOCATION ~IN
DENSITY ALTITUDE ~FT
ROTOR SPEED ~RPM
PROP ANGLE ~DEG
WHEEL HEIGHT ~FT
CONFIGURATION
20540
293.1(FWD)
700
249
-2.2
5
EXT. STORES
GEAR DOWN
TAIL ROTOR POWER ~SHP
TAIL ROTOR POWER UNAVAILABLE
LONGITUDINAL STICK POS ~IN FROM CENTER
FWD
AFT
LATERAL STICK POS ~IN FROM CENTER
LT
RT
PEDAL POS ~IN FROM FULL LEFT
LT
RT
30
20
10
0
10
20
30
LEFT FORWARD (45°)
RIGHT REARWARD (45°)
TRUE AIRSPEED ~ KNOTS

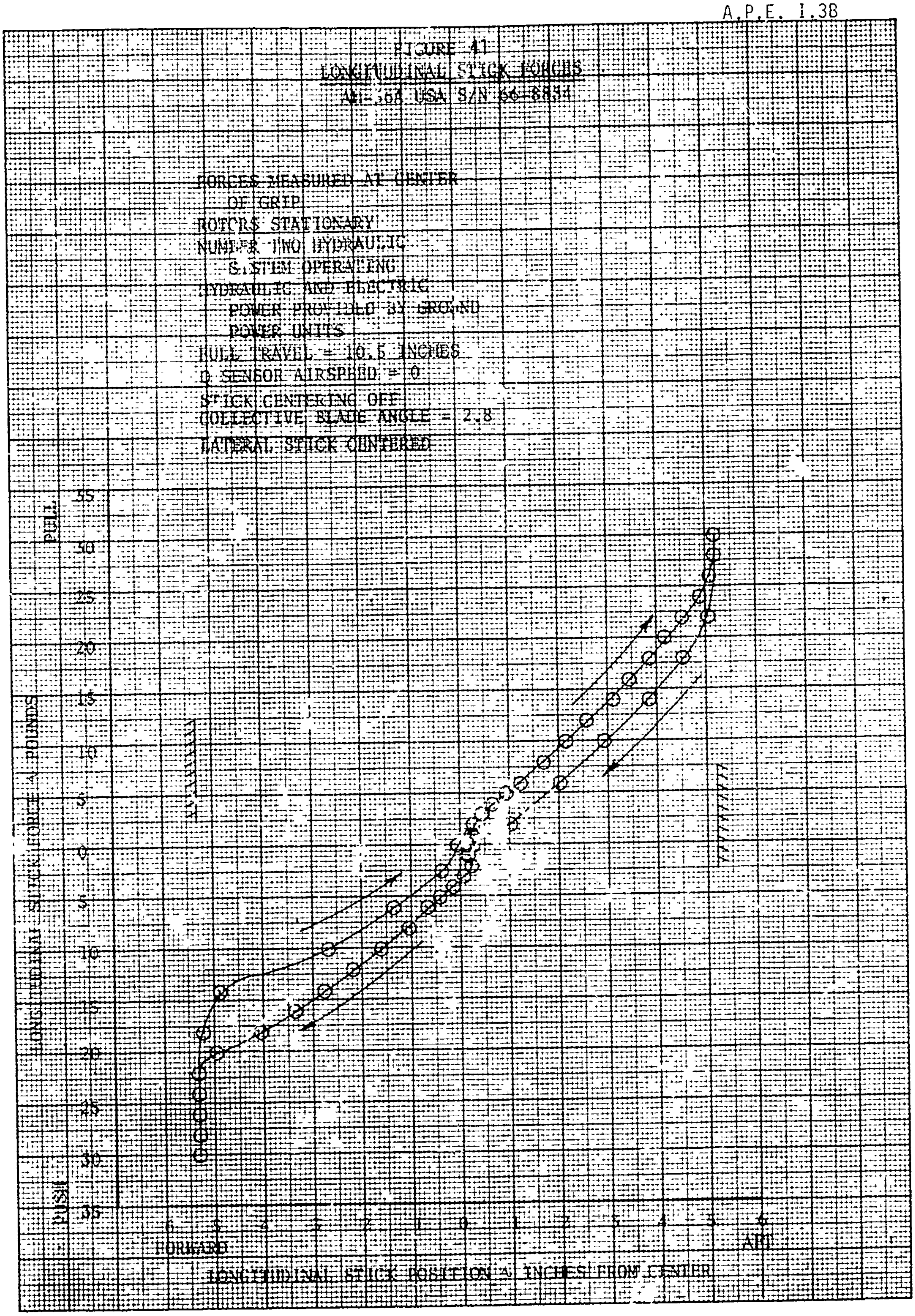
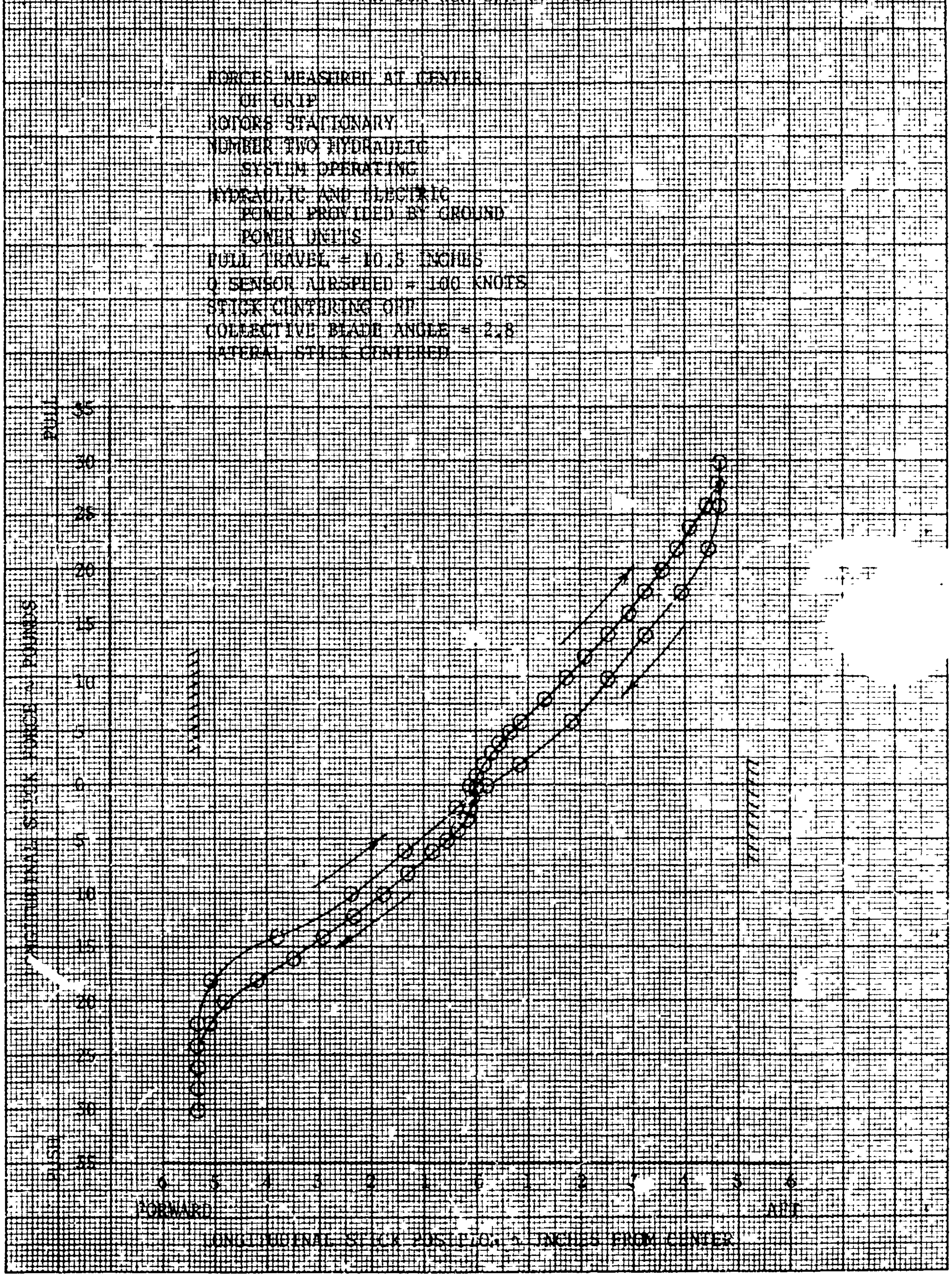


FIGURE 41
LONGITUDINAL STICK FORCES
AH-56A USA S/N 66-8831
FORCES MEASURED AT CENTER
OF GRIP
ROTORS STATIONARY
NUMBER TWO HYDRAULIC
SYSTEM OPERATING
HYDRAULIC AND ELECTRIC
POWER PROVIDED BY GROUND
POWER UNITS
FULL TRAVEL = 10.5 INCHES
Q SENSOR AIRSPEED = 0
STICK CENTERING OFF
COLLECTIVE BLADE ANGLE = 2.8
LATERAL STICK CENTERED
PULL
PUSH
LONGITUDINAL STICK FORCE ~ POUNDS
FORWARD
AFT
LONGITUDINAL STICK POSITION ~ INCHES FROM CENTER

FIGURE 42
LONGITUDINAL STICK FORCES
AH-56A USA S/N 66-8834



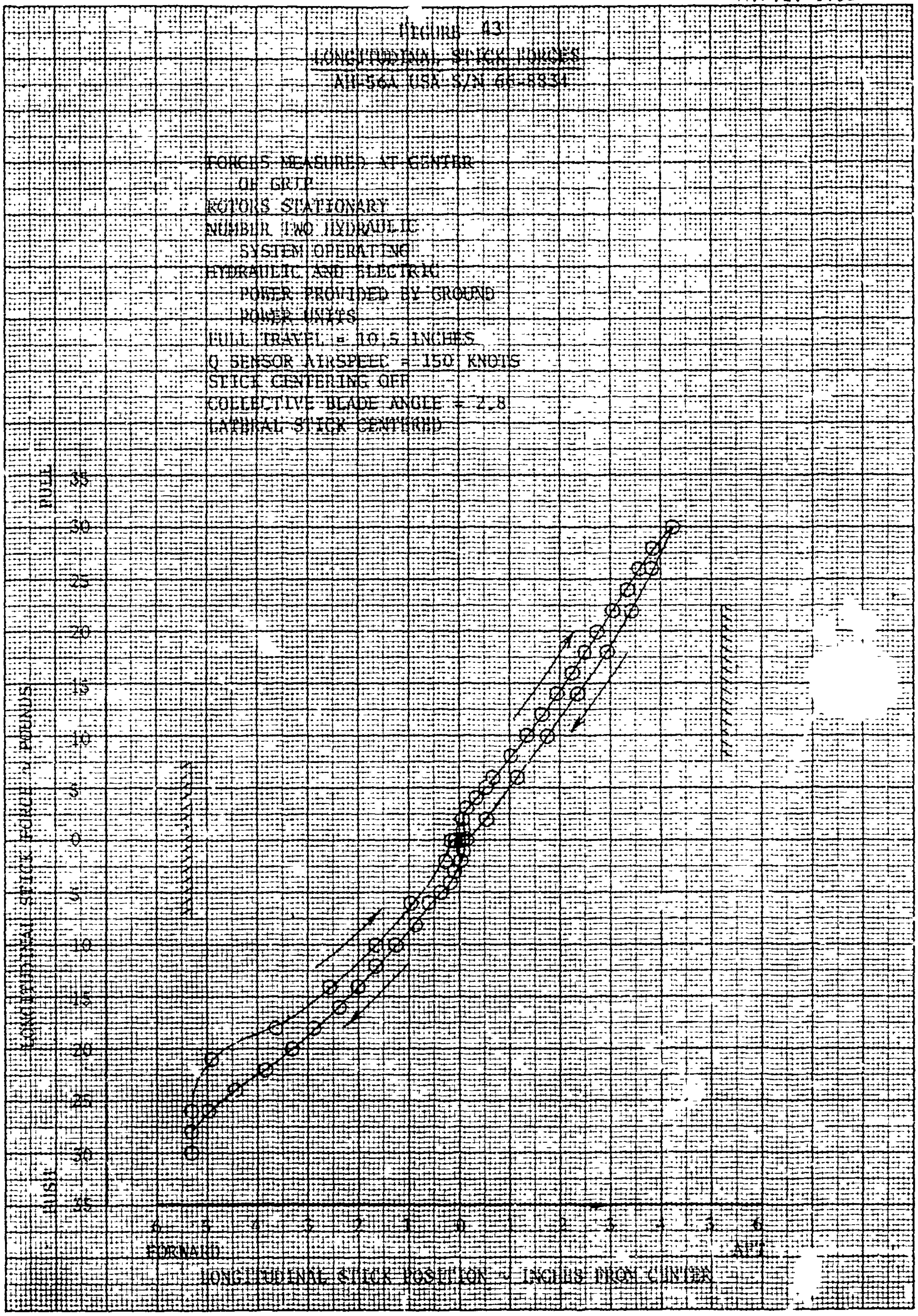


FIGURE 43
LONGITUDINAL STICK FORCES
AH-56A USA S/N 66-8834

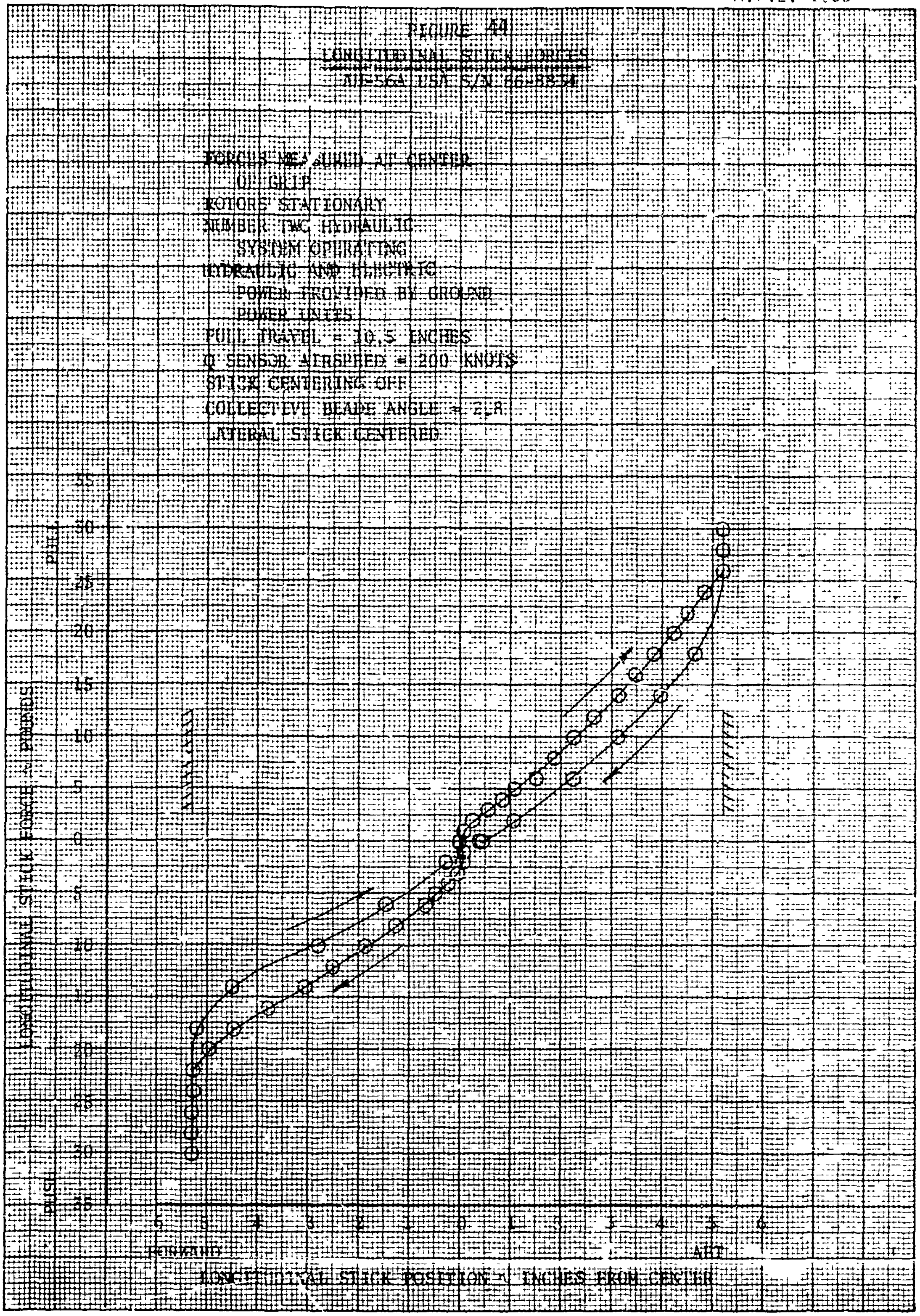


FIGURE 44
LONGITUDINAL STICK FORCES
AH-56A USA S/N 66-8834
FORCES MEASURED AT CENTER OF GRIP
ROTORS STATIONARY
NUMBER TWO HYDRAULIC SYSTEM OPERATING
HYDRAULIC AND ELECTRIC POWER PROVIDED BY GROUND POWER UNITS
FULL TRAVEL = 10.5 INCHES
Q SENSOR AIRSPEED = 200 KNOTS
STICK CENTERING OFF
COLLECTIVE BLADE ANGLE = 2.8
LATERAL STICK CENTERED
PULL
PUSH
LONGITUDINAL STICK FORCE ~ POUNDS
FORWARD
AFT
LONGITUDINAL STICK POSITION ~ INCHES FROM CENTER

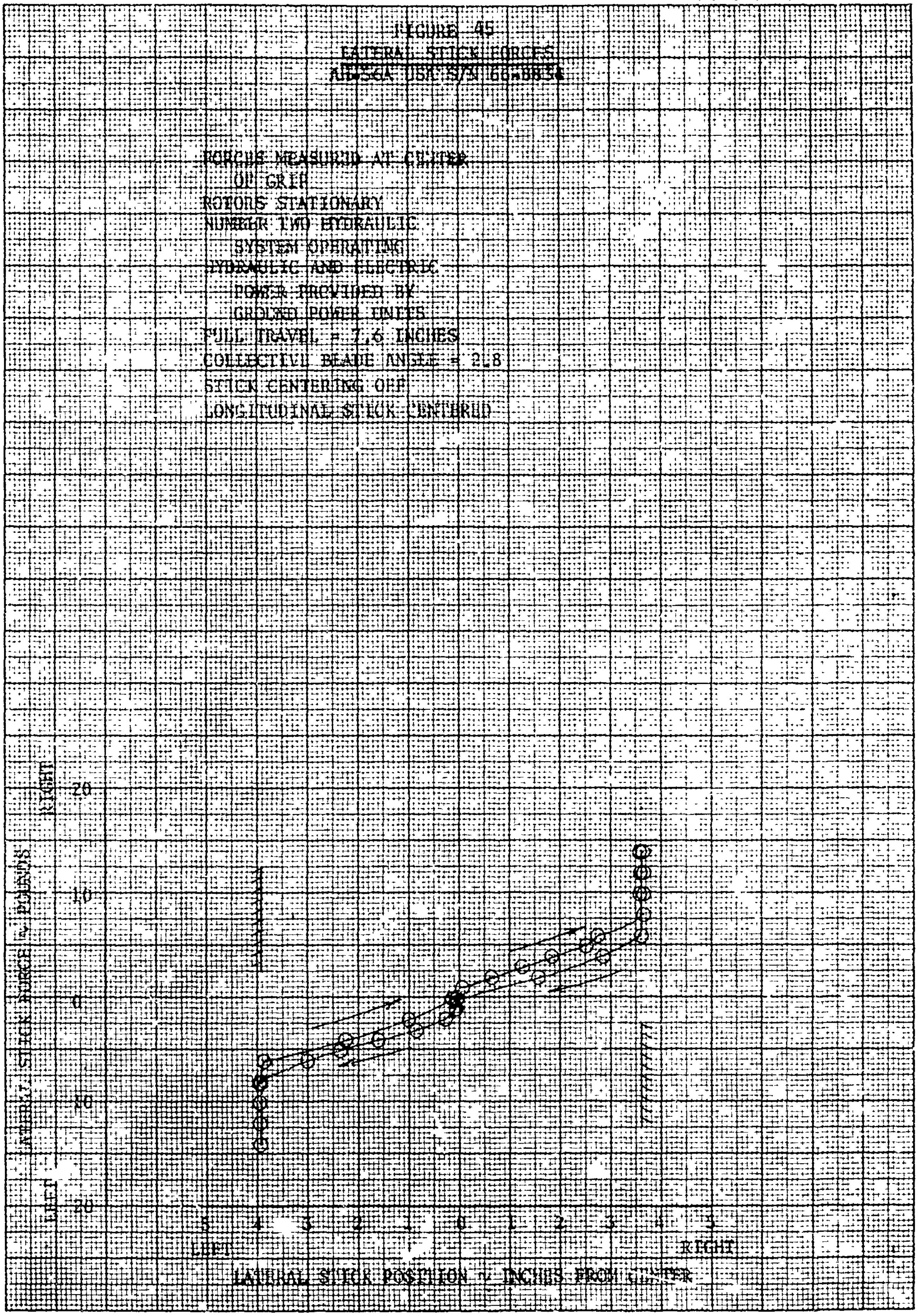


FIGURE 45
LATERAL STICK FORCES
AH-56A USA S/N 66-8834

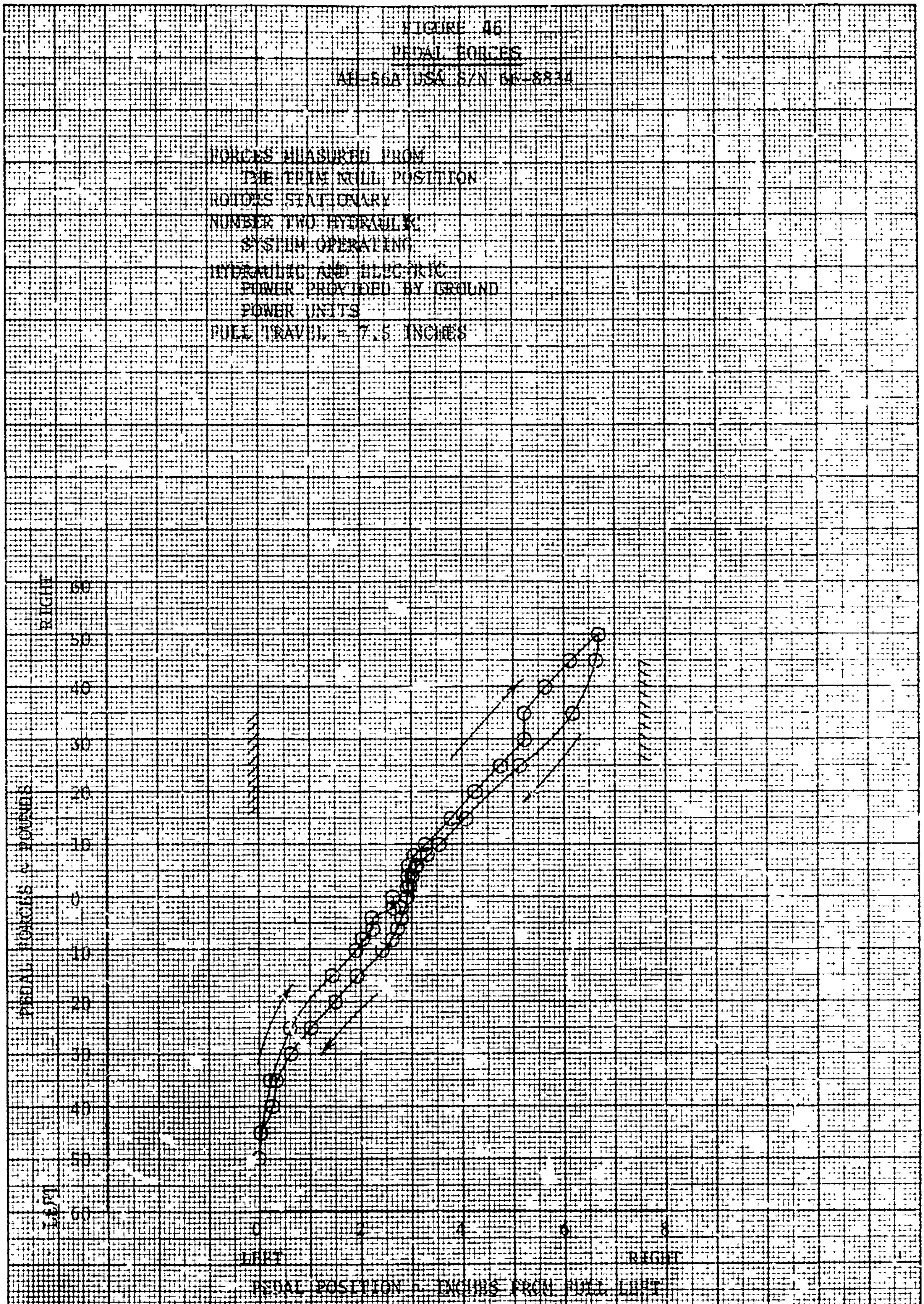


FIGURE 46
PEDAL FORCES
AH-56A USA S/N 66-8834
FORCES MEASURED FROM THE TRIM NULL POSITION
ROTORS STATIONARY
NUMBER TWO HYDRAULIC SYSTEM OPERATING
HYDRAULIC AND ELECTRIC POWER PROVIDED BY GROUND POWER UNITS
FULL TRAVEL = 7.5 INCHES
PEDAL FORCES ~ POUNDS
RIGHT
LEFT
60
50
40
30
20
10
0
10
20
30
40
50
60
0
2
4
6
8
LEFT
RIGHT
PEDAL POSITION ~ INCHES FROM FULL LEFT

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5. AUTHOR(S) (First name, middle initial, last name)

FLOYD L. DOMINICK JR., Project Engineer
GARY L. BENDER, Project Engineer
RANDY D. McCLELLAN, Engineer
GARY C. HALL, LTC, TC, US Army, Project Officer/Pilot
PAUL G. STRINGER, MAJ, CE, US Army, Project Pilot
MARVIN W. BUSS, Project Pilot

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13. ABSTRACT

The Army Preliminary Evaluation I and a portion of the Research and Development Acceptance Test I were conducted periodically on the AH-56A Cheyenne compound helicopter. ~~by the~~ US Army Aviation Systems Test Activity between 30 January and 23 December 1971. These engineering tests were divided into five distinct phases to permit ~~Army evaluation~~ of the aircraft at various stages of the contractor development program. Primary test objectives were to gather stability and control data to provide an early assessment of the AH-56A, to assist in determining flight envelopes for future Army tests, and to examine previously identified problem areas. ~~None~~ of the rotor dynamic instabilities previously encountered in the contractor's development program were noted during these tests. Lateral control migration with airspeed was not objectionable. The capability of the pusher propeller to provide rapid deceleration and to control airspeed independently of dive angle is an excellent feature. Five deficiencies and 54 shortcomings were identified. The deficiencies are (1) excessive pilot workload due to unacceptable static lateral-directional stability characteristics at low airspeed seriously impairs the capability to operate at minimum altitudes unaffected by conditions of darkness or adverse weather, (2) uncommanded aircraft motion and loss of control during some maneuvering flight conditions, (3) rapid rate of rotor speed decay following simulated engine failures which allows the rotor speed to drop below the present transient limit, (4) inadequate directional control margins in sideward flight, and (5) excessive vibration levels in portions of the flight envelope. Correction of the deficiencies should be a prerequisite for an airworthiness release for operational Army aviators, and correction of the shortcomings is desirable. Two deficiencies warrant a reduction of the flight ~~envelope size for future Army tests until~~ correction of those deficiencies is accomplished. Further testing of the AH-56A is recommended.

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Flight envelopes						
Five deficiencies and 54 shortcomings						