AD-771 914

ATTACK HELICOPTER EVALUATION AH-56A CHEYENNE COMPOUND HELICOPTER

John N. Johnson, et al

Army Aviation Systems Test Activity
Edwards Air Force Base, California

June 1972

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RDTE PROJECT NO.
AVSCOM PROJECT NO. 72-08
USAASTA PROJECT NO. 72-08

ATTACK HELICOPTER EVALUATION

AH-56A CHEYENNE COMPOUND HELICOPTER

FINAL REPORT

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

ABSTRACT

The US Army Aviation Systems Test Activity (USAASTA) conducted an Attack Helicopter Evaluation of the AH-56A Cheyenne Compound Helicopter during the period 15 April to 15 June 1972. The AH-56A was tested at Yuma Proving Ground, Arizona and Mammoth Lakes, California. This evaluation was conducted to provide data for use in determining Advanced Aerial Fire Support System effectiveness model inputs, validating material need requirements, and validating contractor claims. The forward area concealment evaluation was conducted by the US Army Combat Developments Command Aviation Agency and the maintenance characteristics evaluation was conducted by the US Army Aviation Systems Command. The performance and handling qualities testing consisted of 49 test flights totaling 42.2 flight hours. The pusher propeller was a major contributor to several enhancing performance and handling qualities characteristics. The results of this evaluation revealed two deficiencies and 24 shortcomings. The deficiencies found during this evaluation were loss of aircraft control, within the flight envelope, resulting from blade moment stall and the inability to effectively perform slow-speed low-level mission tasks below 120 KCAS under adverse weather conditions, due to the combined effects of the lateral-directional control system and static lateral-directional stability characteristics. The shortcomings of most concern were lack of adequate sideforce below 120 KCAS, excessive pitch due to sideslip, a long pitch time constant at high airspeeds, and the high pilot workload required for power management when operating near maximum power.

AD 771914

Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author)
US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

2a. REPORT SECURITY CLASSIFICATION
UNCLASSIFIED

2b. GROUP

3. REPORT TITLE
ATTACK HELICOPTER EVALUATION
AH-56A CHEYENNE COMPOUND HELICOPTER

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)
FINAL REPORT

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6. REPORT DATE
JULY 1972

7a. TOTAL NO. OF PAGES

7b. NO. OF REFS

8a. CONTRACT OR GRANT NO.

b. PROJECT NO.
AVSCOM PROJECT NO. 72-08

c.

d.

9a. ORIGINATOR'S REPORT NUMBER(S)
USAASTA PROJECT NO. 72-08

9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)

10. DISTRIBUTION STATEMENT
Distribut... ...y; test and evaluation, June 1972.
Other re... to the Commanding General,
USAAVSCC... s, Missouri 63166

Approved for public release, distribution unlimited.

11. SUPPLEMENTARY NOTES

12. SPONSORING MILITARY ACTIVITY
US ARMY AVIATION SYSTEMS COMMAND
ATTN: AMSAV-EF
PO BOX 209, ST. LOUIS, MISSOURI 63166

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DD FORM 1 NOV 65 1473 REPLACES DD FORM 1473, 1 JAN 64, WHICH IS OBSOLETE FOR ARMY USE.

UNCLASSIFIED
Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Attack Helicopter Evaluation						
validating requirements and contractor claims						
pusher propeller						
enhancing performance						
two deficiencies						
loss of control						
inability to perform slow-speed, low-level tasks						
24 shortcomings						

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INTRODUCTION

BACKGROUND

1. The AH-56A Cheyenne advanced aerial fire support system was developed by Lockheed Aircraft Corporation (LAC) under contract to the US Army Aviation Systems Command (AVSCOM). Design and development was performed by LAC at its Lockheed-California Company (LCC) facility in Van Nuys, California. The AH-56A was designed specifically to meet an Army Qualitative Materiel Requirement (QMR) dated 17 December 1965. The aircraft was first flown in September 1967. Since that time, many developmental modifications have been made to the basic aircraft, most notably to the main rotor and flight control systems. The US Army Aviation Systems Test Activity (USAASTA) previously conducted an Army Preliminary Evaluation (APE) and a Research and Development Acceptance Test (RDAT) of the AH-56A helicopter (ref 1, app A). The USAASTA was tasked by AVSCOM (ref 2) to conduct the attack helicopter evaluation to support the Attack Helicopter Requirements Evaluation being conducted by the US Army Combat Developments Command.

TEST OBJECTIVES

2. The objectives of the AH-56A attack helicopter evaluation were as follows:

a. To provide data for use in determining Advanced Aerial Fire Support System (AAFSS) effectiveness model inputs.

b. To provide data for validating materiel need (MN) requirements.

c. To provide data for validating contractor claims.

DESCRIPTION

3. The AH-56A is a compound helicopter designed to perform the AAFSS mission. In addition to a single main hingeless rotor and a conventional antitorque tail rotor, a pusher propeller is located at the aft end of the fuselage, and a wing is located low on the midsection of the fuselage. The design allows the main rotor to be partially unloaded in high-speed forward flight. Potential exists for installation of a variety of armament systems in the two turrets and on six

Table 1. Test Conditions.[1]

Type of Test	Nominal Gross Weight (lb) Clean[2]	Stores[2]	Nominal Density Altitude (ft)	Nominal Trim Calibrated Airspeed (KCAS)
Hover performance[3]	17,800	19,500	1500 8000	0
Level flight performance	18,000	18,800	6500 4850	100 to 186
Acceleration and deceleration performance		19,200	1500	N/A
Lateral acceleration performance and agility		18,750	1200	N/A
Takeoff and landing	18,500	19,700	1000 8000	N/A
Sideward and rearward flight[4]		18,500	1200 8250	0 to 25 0 to 15
Control positions in trimmed forward flight	17,100 to 18,500	18,100 to 19,000	5000 10000	100 to 186
Trimmability	17,100 to 18,800	18,100 to 20,500	1500 8000	0 to 199
Static longitudinal stability		20,500	4000	120, 150, 190
Static lateral-directional stability		19,500	5000	60, 120, 150 190
Dynamic stability		19,800	4500	120, 150
Controllability		19,800	4500	120, 150
Maneuvering stability	17,050	20,000	4600 7300	120, 150, 190 80
Simulated engine failure		20,100	2700	60 to 190
Autorotational characteristics		19,700	2700	95
Automatic stabilization system characteristics	18,500	20,000	2000 6000 8000	0 to 170
Typical mission maneuvers[5]		18,300 to 19,700	1000 5000	0 to 180

[1] Rotor speed: 246 rpm (100 percent; cg range FS 299.0 to 299.8 (aft); combinations of weights, configurations, and speeds.
[2] Clean (no external stores); stores (two XM159 pods on each wing).
[3] In-ground-effect (10 foot wheel height), out-of-ground effect; rotor speed 95 to 105 percent.
[4] Center-of-gravity (cg) at FS 296.4 (mid).
[5] Dives, pop-up, simulated TOW launches and tracking maneuvers, and rolling pull-ups.

external stores stations. The landing gear is of the conventional retractable wheel type. The AH-56A is powered by a single General Electric T64-GE-716(ST) turboshaft engine which has a maximum power rating of 4330 shaft horsepower (shp) at sea-level (SL), standard-day conditions. The aircraft transmission is limited to 3925 shp. A general aircraft description, flight control system description, and photographs are contained in appendixes B, C, and D, respectively.

SCOPE OF TEST

4. The majority of the testing was conducted at Yuma Proving Ground, Arizona (elevation 400 feet), with high altitude testing conducted at Mammoth Lakes Airport, California (elevation 7132 feet). During this test program, 49 test flights were conducted for a total of 42.2 flight hours. The test conditions are shown in table 1.

5. The AH-56A was tested for compliance with the applicable paragraphs of military specifications MIL-H-8501A (ref 3, app A) and MIL-F-8785(ASG) (ref 4). The flight restrictions and limitations are contained in the operator's manual (ref 5) and the safety-of-flight release (app E).

METHODS OF TEST

6. Standard engineering flight test methods were used and are briefly described in the Results and Discussion section of this report. Test results were compared to the applicable requirements of the military specifications (refs 3 and 4, app A). A Handling Qualities Rating Scale (HQRS) (app F) was used during evaluation of mission tasks. Data analysis methods are described in appendix G.

7. The test aircraft was equipped with calibrated instrumentation which was installed and maintained by the contractor. A detailed list of the test instrumentation is presented in appendix H.

CHRONOLOGY

8. The chronology of the AH-56A attack helicopter evaluation is as follows:

Test request received	13 March 1972
Testing commenced at Yuma, Arizona	15 April 1972
Test completed at Yuma, Arizona	30 May 1972
Testing commenced at Mammoth Lakes, California	5 June 1972
Testing completed	15 June 1972

RESULTS AND DISCUSSION

GENERAL

9. An evaluation of performance and handling qualities of the AH-56A was conducted to provide data for use in determining Advanced Aerial Fire Support System effectiveness model inputs, validating material need requirements, and validating contractor claims. The pusher propeller was a major contributor to several enhancing performance and handling qualities characteristics. The results of this evaluation revealed two deficiencies and 24 shortcomings. The deficiencies found during the evaluation were loss of aircraft control, within the flight envelope, resulting from blade moment stall and the inability to effectively perform slow-speed, low-level mission tasks below 120 KCAS under adverse weather conditions due to the combined effect of the lateral-directional control system and static lateral-directional stability characteristics. The shortcomings of most concern were lack of adequate sideforce below 120 KCAS, excessive pitch due to sideslip, a long pitch time constant at high airspeeds, and the high pilot workload required for power management when operating near maximum power.

PERFORMANCE

General

10. Performance testing of the AH-56A Cheyenne compound helicopter included hover and level flight performance, forward-flight acceleration and deceleration performance, and lateral acceleration performance. The standard day in-ground-effect (IGE) and out-of-ground (OGE) hover ceilings for the current maximum gross weight (20,500 pounds) are approximately 9300 and 6000 feet, respectively. The cruise speed and maximum speed in level flight (V_H) at sea level for the TOW-mission configuration (18,750 pounds gross weight) are 158 and 194 KTAS, respectively. The acceleration performance was excellent to 120 KTAS and satisfactory to V_H; however, maximum power could not be used throughout the maneuver because of the high pilot workload required for power management. The deceleration performance was outstanding. The highest average lateral acceleration to 30 KTAS was 0.27g in right sideward flight and 0.20g in left sideward flight.

Hover Performance

11. In-ground-effect and OGE hover testing was accomplished at Yuma Proving Ground, Yuma, Arizona (400 feet, mean sea-level (MSL)) and Mammoth Lakes Airport, California (7132 feet, MSL). The free-flight hover technique was used for both IGE (10 foot wheel height) and OGE testing. The aircraft was tested at gross weights ranging from 16,900 to 20,300 pounds with a propeller blade angle of -2.2 degrees (hover detent). Propeller blade angles other than this value changes the power required to hover and affects hover ceilings as well as aircraft attitudes. The hover performance of the AH-56A are summarized in figures 1 and 2, appendix I. The maximum hover performance is limited by the current operating envelope. Nondimensional summaries of aircraft hover performance and tail rotor performance are presented in figures 3 through 6. The standard-day hover ceiling at the current maximum gross weight (20,500 pounds) is approximately 9300 feet IGE and 6000 feet OGE. The 95°F day hover ceiling at maximum gross weight is 4650 feet IGE and 1500 feet OGE.

12. The IGE and OGE hover performance at the CDCEC 43.6/IV test gross weight of 18,750 pounds (app J) is presented in figure A. At 18,750 pounds the IGE and OGE hover ceilings are 11,800 and 8,500 feet, respectively, on a standard day and 6800 and 3850 feet, respectively on a 95°F day.

Level Flight Performance

13. Level flight performance tests were conducted using the constant coefficient of thrust (C_T) method described in appendix G. The tests were accomplished to determine the variation of power required with airspeed at standard rotor speed at the conditions listed in table 2. The results of the tests are presented in figures 7 through 13, appendix I.

14. Within the airspeed range tested, increasing collective blade angle resulted in a reduction of power required (figs. 11 and 12, app I). As shown in figure B, the addition of external stores caused an increase in equivalent flat plate area (ΔF_e) of approximately two square feet at 110 KTAS increasing to six square feet at 200 KTAS. By extrapolating the data obtained in the clean configuration, applying a drag correction for external stores, and assuming the equivalent drag of XM159 rocket pods being equal to the equivalent drag of TOW missile pods, a speed-power polar at a collective blade angle of 5 degrees was obtained for the TOW-mission configuration at sea-level, standard-day conditions (fig. 13). These data show a cruise speed of 158 KTAS and a maximum speed in level flight of 194 KTAS at 3925 shp, the transmission limit which is below maximum continuous (normal rated) power at sea level.

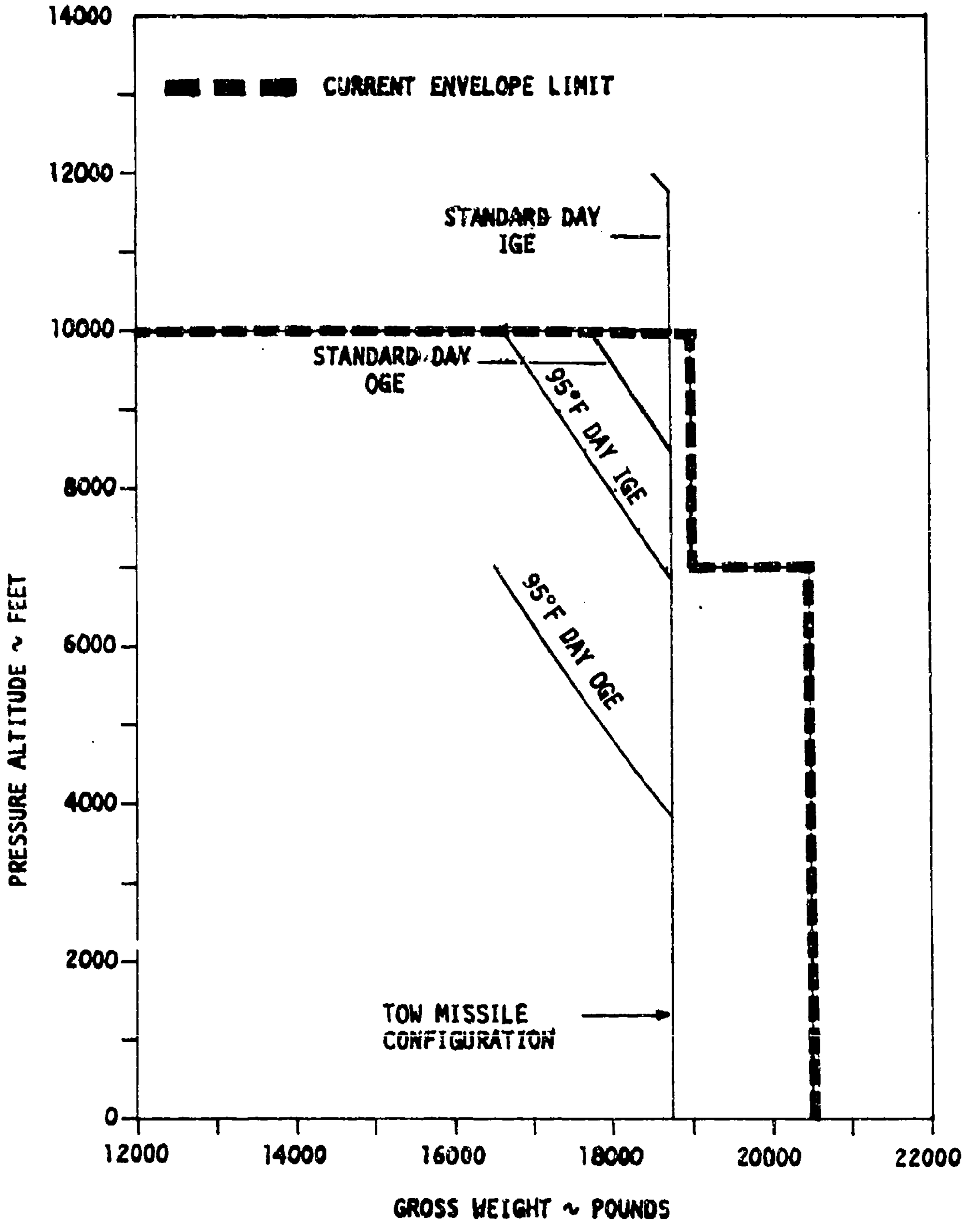


FIGURE A. HOVER PERFORMANCE SUMMARY
TOW MISSILE CONFIGURATION.

Table 2. Level Flight Performance Test Conditions.

Referred[1] Gross Weight (lb)	Configuration	Collective Blade Angle (deg)
19,830	Clean	5
21,720	Clean	5
21,720	Ext Stores	3
21,720	Ext Stores	5
21,720	Ext Stores	7
23,970	Clean	5

[1] $\frac{\text{Weight}}{\text{Density ratio}} \left(\frac{\omega}{\sigma}\right)$

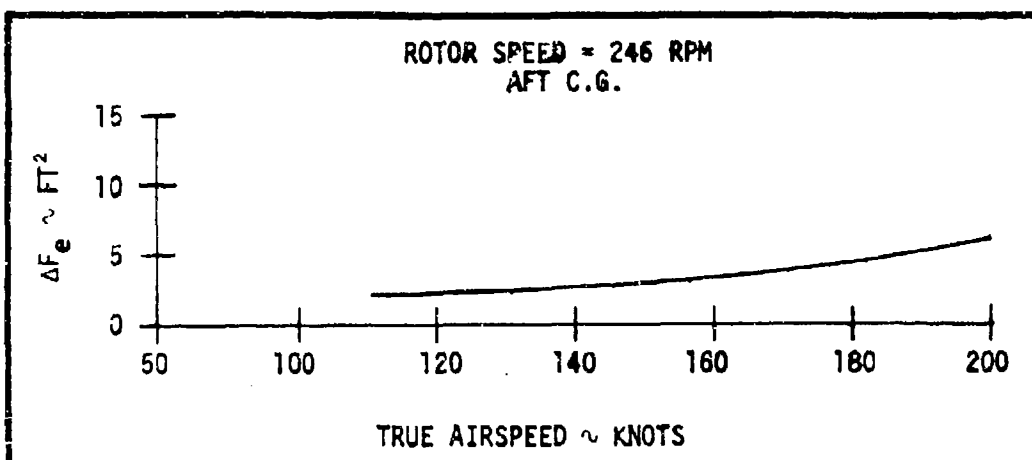


FIGURE 8.
CHANGE IN EQUIVALENT FLAT PLATE AREA (ΔF_e)
BETWEEN CLEAN AND EXT STORES CONFIGURATIONS

Forward Flight Acceleration and Deceleration Performance

15. Maximum forward acceleration performance was determined by accelerating from a hover to V_H at constant altitude while maintaining near maximum power. Starting from a hover, propeller pitch was increased from the hover detent (-2.2 degrees) to approximately +10 degrees. Simultaneously, the aircraft was pitched nose low and the collective blade angle was increased until maximum power was reached. As the aircraft accelerated, maximum power was maintained by increasing propeller pitch while following the maximum collective schedule (app E). The landing gear was raised at approximately 95 KCAS. Maximum deceleration performance was determined by decelerating from V_H to hover at constant altitude. Starting at V_H at 5 degrees collective, the deceleration was begun by decreasing propeller pitch to -12 degrees (current full reverse position) in a smooth motion at a moderate rate. The landing gear was lowered at approximately 120 KCAS. Longitudinal cyclic was used to maintain a constant altitude until the aircraft approached a hover and collective was increased. At a hover propeller pitch was increased to -2.2 degrees. The tests were flown at the following conditions: nominal gross weight of 19,200 pounds; aft cg; external stores installed; 1500-foot density altitude. The average accelerations and decelerations between several airspeeds are presented in table 3. Aircraft attitude was easily controlled by the pilot during these tests; however, the power management workload was high and, as a result, maximum power could not be maintained throughout the accelerations (fig. 14, app I). The acceleration performance is excellent from hover to approximately 120 KTAS and satisfactory to V_H. The aircraft was consistently decelerated from 196 KTAS to a hover in less than 35 seconds within 4000 feet of stopping distance (fig. 15). The deceleration performance of the AH-56A is outstanding.

Table 3. Forward Acceleration and Deceleration Performance.

Test	Airspeed[1] Range (KTAS)	Average Acceleration (g)	Time (sec)
Acceleration	0 to 188	0.09	107.0
Acceleration	165 to 188	0.02	54.0
Deceleration	196 to 0	0.31	33.0
Deceleration	196 to 165	0.18	5.4

[1] Deceleration started from 8 knots above dash speed.

Lateral Acceleration Performance

16. The lateral acceleration performance was determined by accelerating from a stabilized hover (approximately 50 feet) to the envelope limit. This maneuver was accomplished by establishing a target bank angle with rapid lateral control inputs while simultaneously adjusting collective to maintain constant altitude during the acceleration. The target bank angle was increased on successive maneuvers until maximum acceleration performance was achieved. The propeller pitch was held constant at -2.2 degrees. Performance data were recorded by ground operated space positioning equipment. These tests were accomplished in the external stores configuration at the TOW mission gross weight of 18,750 pounds and ar enter-of-gravity (cg). Density altitude was 1200 feet and s ace winds were less than 3 knots. Results of these tests are shown in table 4 and figure 16, appendix I.

Table 4. Maximum Lateral Acceleration Performance.

Direction of Flight	Bank Angle (deg)	Maximum Acceleration (g)	Average Acceleration to 30 KTAS (g)	Time to 30 KTAS (sec)
	12.3	0.12	0.11	14.4
Left	17.3	0.20	0.16	10.0
	20.5	0.30	0.20	7.9
	10.7	0.23	0.17	9.4
Right	20.3	0.34	0.21	7.4
	18.7	0.34	0.27	5.8

HANDLING QUALITIES

General

17. Static and dynamic stability and control tests were conducted to qualitatively and quantitatively evaluate the handling qualities of the AH-56A Cheyenne compound helicopter. In addition, the

helicopter response characteristics following simulated failures of the engine and of the automatic stabilization systems were separately evaluated. The capability of using propeller pitch during slope landings and takeoffs or hovering to attain a desired aircraft attitude is an enhancing feature. Results of the handling qualities evaluation reveal one deficiency and 13 shortcomings. The deficiency is loss of aircraft control within the flight envelope resulting from blade moment stall. The shortcomings include lack of adequate side force at airspeeds below 120 KCAS, excessive pitch due to sideslip and the long pitch time constant at high airspeeds. Power management requires an undue amount of the pilot attention during high power operation or when the power margin is small.

Control Systems Characteristics

18. In contrast to the normally configured AH-56A helicopter which has the pilot station in the aft cockpit, the test aircraft was configured with the pilot station in the forward cockpit. The front cockpit control breakout forces, force gradients, and ranges of travel were determined during ground tests with the rotor stationary and auxiliary power unit (APU) operating. Only the number-two hydraulic system was pressurized during APU operation. Control forces were measured from the center of the cyclic grip, the base of the directional pedals, and at the center of the propeller control twist grip on the collective lever. Breakout forces (including friction) were determined by recording the forces required to obtain initial movement of each control. Data from these tests are presented in figures 17 through 21, appendix I, and summarized in table 5.

19. A restriction in lateral control travel was present when full lateral trim was used. With full right lateral trim the left lateral control travel was reduced 0.6 inches and with full left lateral trim the right lateral control travel was reduced 0.4 inches. Adequate lateral control authority remained to safely fly the aircraft in case of a runaway lateral trim motor. The 3.2 pound per inch longitudinal control force gradient failed to meet the requirement of paragraph 3.2.4 of MIL-H-8501A. The average directional control force gradient was 13 pounds per inch. Large directional breakout and friction forces were also apparent to the pilot and not in harmony with the lateral and longitudinal control forces. This characteristic contributed to the considerable pilot effort required to maintain coordinated flight (HQRS 5). The 8-pound directional breakout force to the right and the high limit directional control force did not meet the requirements of paragraph 3.3.13 of MIL-H-8501A as measured on the ground. The excessive breakout force in the directional control pedals and the excessive directional control force gradients are shortcomings which should be corrected.

Table 5. Control Breakout Forces and Force Gradients.

Test	Allowed By MIL-H-8501A Minimum	Allowed By MIL-H-8501A Maximum	Test Results
Longitudinal breakout	0.5	1.5 lb	1.0 lb aft, 1.5 lb fwd
Longitudinal breakout[1]	N/A	N/A	2.0 lb aft, 2.0 lb fwd
Longitudinal gradient	0.5 lb/in.	2.0 lb/in.	3.2 lb/in.
Longitudinal gradient[2]	N/A	N/A	4.5 lb/in.
Lateral breakout	0.5 lb	1.5 lb	1.0 lb
Lateral gradient	0.5 lb/in.	2.0 lb/in.	1.4 lb/in.
Pedal breakout	3.0 lb	7.0 lb	8.0 lb right, 7.0 lb left
Pedal gradient	N/A	N/A	13.0 lb/in.
Collective breakout[2]	N/A	N/A	7.5 lb
Collective breakout[3]	N/A	N/A	18.0 lb

[1] Dynamic pressurized ("Q") sensor pressurized to equivalent of 190 KCAS.
[2] Electric friction ON; mechanical friction minimum.
[3] Electric friction ON; mechanical friction maximum.

Takeoff and Landing Characteristics

20. Takeoff and landing characteristics were evaluated in the clean configuration at weights up to 19,700 pounds and in the external stores configuration to 20,500 pounds in winds up to approximately 15 knots. The evaluation included lift-off to hover and touchdown from a hover on level surfaces and on slopes to 11.9 degrees, normal hover takeoffs and landings, maximum acceleration and deceleration hover takeoffs and landings, and rolling takeoffs and landings.

21. Lift-off to and touchdowns from a hover were smooth with positive attitude control and minimal pilot workload. The longitudinal and lateral controls cannot be pretrimmed on the ground due to a trim null system which is activated when the landing gear oleos are compressed. There were no mechanical instabilities observed when light on the gear or when moving the collective control slowly or rapidly. There was no noticeable change in handling qualities on lift-off or touchdown with the roll compensator OFF. Maintaining the aircraft position on the ground was easily accomplished using either propeller thrust or wheel brakes. The initial part of the evaluation was flown on standard landing gear and the gear struts failed to compress evenly as the collective was lowered. Instrumented gear with improved struts was installed on 19 May 1972, prior to the slope landing tests. These struts compressed much more evenly.

22. The slope landing and takeoff evaluation was conducted in the clean configuration at an average gross weight of 17,760 pounds and in the external stores configuration (76 rockets) at an average gross weight of 20,150 pounds. Wind speed was from 4 to 8 knots cross slope (right to left as viewed from the base of the slope) for both configurations. The desert surface was coated with asphalt oil to reduce blowing dust. Each of the landings was commenced from a stabilized hover with the lateral and longitudinal trim centered to prevent cyclic inputs to the rotor if the trim null system failed to actuate. The point was considered valid if, after landing, the collective could be lowered to the pneumatic-down stop (3.2 degrees) and the cyclic centered (nulled). At each point the pilot attempted to hold the aircraft on the slope with brakes only (zero propeller thrust). The test results are presented in table 6.

23. For vertical ascents and descents on cross slopes, up to 1 1/2 inches of lateral cyclic was required. Prior to these tests, the contractor demonstrated that inputs of 2-inch lateral cyclic did not produce excessive main rotor shaft bending moments. The aircraft attitude was uncomfortable beyond a 10-degree roll attitude but the aircraft was stable to the limits tested. One half-inch lateral control inputs were made in both directions at each slope angle

tested with no adverse aircraft response or excessive loads. Takeoffs were also evaluated perpendicular to the slope but these resulted in a downslope translation which was very uncomfortable. The vertical ascent is preferred.

24. Upslope landings and takeoffs were accomplished from a level attitude using zero thrust on the propeller to slopes of approximately 7 degrees. Above a 7-degree upslope, and for all downslope landings, the aircraft attitude was matched to the slope by using propeller pitch and longitudinal cyclic. This method was preferred because vertical descents and ascents were easily achieved and the tendency to use large longitudinal control inputs was significantly reduced. The contractor demonstrated 2-inch longitudinal control inputs without excessive main rotor shaft bending moment. Less than 1/2 inch of longitudinal control was required for this test. The maximum aircraft attitudes attained up and downslope were not uncomfortable. The aircraft could be held with brakes only (zero thrust on propeller) except for upslopes which exceeded 9 degrees. At slopes greater than 9 degrees, the landing gear slid through the surface material. All of the slope landings and takeoffs were easily made. The capability of using the propeller pitch during slope landings and takeoffs or hovering to attain a desired aircraft attitude is excellent.

Table 6. Slope Landing Test Results.

Configuration	Aircraft Attitude (deg)	Slope Angle (deg)
Clean	13.7 Nose up	11.4
Ext stores	12.9 Nose up	11.9
Clean	10.4 Nose down	9.8
Ext stores	8.5 Nose down	9.2
Clean	14.7 Right wing low	8.4
Ext stores	15.2 Right wing low	8.2
Clean	15.6 Left wing low	8.2
Ext stores	15.2 Left wing low	9.9

25. Normal transition to forward flight from a hover was accomplished by increasing propeller thrust while maintaining attitude with the cyclic and directional controls and height with collective. During acceleration through translation, large trim changes occurred about all axes. Control inputs were required to preclude the aircraft from rolling right, pitching up, and yawing left. The trim changes, at an approximate gross weight of 19,500 pounds in the external stores configuration, were approximately 1.1 inches left lateral control, 0.7 inches forward longitudinal control, and 2.3 inches right directional control. A time history of a typical transition is presented in figure 22, appendix I. A tendency to overcontrol in the roll axis was present but was easily corrected. When pretrimmed for forward flight (2 units nose down, 1 1/2 units left wing low, and 1 1/2 units right directional), moderate control forces were required in a hover. The trim changes in transition were much less obvious and pilot workload to maintain precise attitude control was greatly reduced, as was the tendency to overcontrol in the roll axis. Starting with zero trim or when trimmed for a hover, attitude control during transition is extremely difficult, and pilot workload was high because of the large trim changes (HQRS 6). The pilot workload in a hovering takeoff was compounded by the power management requirements inherent in two power controls, propeller pitch and main rotor collective pitch, as discussed in paragraph 81. The high pilot workload during hover takeoffs is a shortcoming and should be corrected.

26. Transition from forward flight to a hover presented a similar pilot workload for attitude control but a very light workload for power management. The large trim changes resulted in high control forces, especially directional, unless trimmed out during the deceleration. The slow rate of operation of the directional trim was distracting (HQRS 3). The tendency to overcontrol in the roll axis was present, but less noticeable than on takeoff.

27. Maximum acceleration takeoffs from a hover were made by pretrimming about all axes for forward flight and setting approximately +10 degrees propeller pitch while in the hover. The aircraft was then accelerated by increasing collective to obtain maximum power and pitching nose low to maintain a constant height. As the aircraft accelerated, maximum power was maintained with propeller pitch as collective was lowered to maintain the maximum allowable collective schedule (app E). The landing gear was raised at approximately 95 KCAS. The pilot workload was slightly increased from normal hover takeoff with pretrim but was not excessive.

28. Maximum deceleration approaches to a hover were made by establishing low-level flight at V_H with a collective setting of 5 degrees. The deceleration was initiated by decreasing propeller pitch to -12 degrees (current full reverse position) at a moderate rate. The landing gear was lowered at approximately 120 KCAS. Longitudinal cyclic was used to maintain a constant altitude until near a hover when collective pitch was increased. Propeller pitch was set at -2.2 degrees at termination to a hover. The pilot workload during the maximum deceleration was no greater than during a normal transition from forward flight to hover; however, the vibration level increased significantly when reverse propeller pitch was used through the transition. Within the scope of this test, the level flight acceleration and deceleration characteristics of the AH-56A are satisfactory.

29. Rolling takeoffs and run-on landings were made from a hard-surfaced runway. Rolling takeoffs were made by accelerating the aircraft to a liftoff speed of approximately 60 KIAS using propeller thrust, with the collective at 3 degrees, and then making a small increase in collective pitch to become airborne. The high pilot workload associated with the trim shifts in a hover transition was avoided in a rolling takeoff. Additionally, pretrimming directionally reduced pedal forces and pilot effort during liftoff. At lift-off, a nose-up pitching was experienced which was easily corrected. Power management was easily accomplished. Run-on landings were conducted at speeds up to approximately 65 KIAS and were easier than hover landings. Reverse thrust on the ground was a very effective braking device and was accompanied by a right yaw and right roll which were easily corrected. The rolling takeoff and run-on landing characteristics of the AH-56A are satisfactory.

Sideward and Rearward Flight Characteristics

30. Tests were conducted to determine control margins and handling qualities while hovering in winds. The aircraft was stabilized at 5-knot increments while tracking a calibrated ground pace vehicle. During forward and sideward flight the propeller pitch control was set at the hover detent. During rearward flight reverse thrust was used to maintain a near-level aircraft attitude. These tests were conducted at the conditions shown in table 7.

Table 7. Lowspeed Flight Test Conditions.[1]

Flight Direction	Limit Airspeed (KTAS)	Density Altitude (ft)	Average Gross Weight (lb)
Forward	none		
Rearward	27	1200	18,880
Sideward	31		
Forward	none		
Rearward	15	8250	18,275
Sideward	17		

[1] Center-of-gravity at FS 296.4 (mid). Wheel height 10 feet. External stores configuration. Winds less than 3 knots.

31. Control positions in sideward flight are presented in figures 23 and 25, appendix I. The lateral control trim position variations were minimal but the control excursions in attaining the desired airspeed were considerable. Increasing forward longitudinal control displacement was required with increasing right sideward speed and increasing aft longitudinal control displacement was required with increasing speed in left sideward flight. The directional control position gradient was stable (increasing left pedal for increasing right sideward flight and increasing right pedal for left sideward flight). The pilot workload increased at the higher speeds because of large trim shifts in transition but aircraft control was not in question. Control margin was adequate at all speeds tested but the safety-of-flight release prevented testing to the 35-knot sideward flight requirement of MIL-H-8501A. Inadequate directional control margins were a deficiency during the APE I testing (ref 1, app I). Subsequently, the range of available tail rotor blade angles was changed, and the control margins are now satisfactory. Further testing at maximum gross weight would be required to completely evaluate the sideward flight handling qualities.

32. Control position data in slow-speed forward and rearward flight are presented in figures 24 and 26, appendix I. Pitch attitude was approximately 2 to 3 degrees nose-up throughout this test but could have been varied as desired through use of propeller pitch. The lateral and directional control position changes with changing airspeed were small at a 8250-foot density altitude but the evaluation was conducted below 17 KTAS (envelope limit). Below that airspeed, the longitudinal control gradient was positive (increasing forward longitudinal control displacement for increasing forward speed, increasing aft longitudinal control displacement for increasing rearward speed). A similar gradient was seen at a 1200-foot density altitude except that a reversal of the gradient occurred at a forward airspeed of approximately 10 KTAS. From 10 to 25 KTAS in forward flight the pilot workload increased, as evidenced by discontinuities in trim control positions, but was acceptable. Control margins were adequate at all speeds tested but the safety-of-flight release prevented testing to the 30-knot rearward flight requirement of MIL-H-8501A. Further testing at maximum gross weight would be required to completely evaluate the rearward flight handling qualities.

Lateral Acceleration Handling Qualities

33. The lateral acceleration handling qualities were evaluated during the lateral acceleration performance testing at the conditions outlined in paragraph 16. Additional testing was conducted at 8400 foot density altitude.

34. At 1200-foot density altitude, control of roll attitude during the acceleration required minimal pilot compensation (HQRS 3). The pilot had positive control of roll attitude; however, there was some tendency to overcontrol in roll. The longitudinal and directional trim shifts during the acceleration were large, requiring relatively high control forces. Extensive pilot effort was required to maintain heading within ±5 degrees of the desired heading at 20 KTAS and above (HQRS 6). Very little difference in pilot effort was observed between left and right accelerations. The excessive pilot workload required to maintain desired heading during sideward acceleration at and above 20 KTAS is a shortcoming and should be corrected.

35. During the tests at a 1200-foot density altitude, the sideward acceleration was so high that it was very difficult to use the maximum acceleration capability of the aircraft and still remain inside the sideward airspeed envelope. Maximum decelerations were attempted by changing the bank angle at a moderate rate to the same bank angle in the opposite direction. During one of the maximum performance decelerations from right lateral accelerations,

pilot-coupled roll oscillations occurred and high main rotor blade chord bending loads were encountered. Although the loads did not reach the inspection limit, a precautionary visual inspection was made and no damage was found. A time history of this maneuver is presented in figure 16, appendix I. Pilot coupled roll oscillations and large longitudinal trim shifts resulted in high pilot workload during this reversal. Extensive pilot effort was required to control the aircraft attitude and to maintain altitude (HQRS 6). Other reversals were made with half the opposite bank angle and were mild by comparison, requiring only moderate pilot effort; however, stopping distance was increased accordingly (HQRS 4). The excessive pilot effort required to control aircraft attitude and maintain altitude in a rapid reversal of lateral acceleration is a shortcoming and should be corrected.

36. At an 8400-foot density altitude, the airspeed was restricted to a maximum of 17 KTAS by the safety-of-flight release. During this evaluation, roll attitude of approximately 10 degrees during the acceleration and approximately 14 degrees during the reversals were used. Maximum acceleration capability of the aircraft could not be used because the sideward limit of 17 KTAS would have been exceeded. The shortcomings noted at a 1200-foot density altitude were not seen because of the 17 KTAS envelope limit.

Control Positions in Trimmed Forward Flight

37. Control positions in trimmed forward flight were determined in the clean and external stores configuration at an aft cg. The data were obtained during level flight performance tests with the aircraft stabilized at zero sideslip. Collective blade angle was held fixed for each of the tests. Data were gathered at collective blade angles of 3, 5, and 7 degrees. Gross weight in the clean configuration varied from 17,000 to 18,500 pounds and for the external stores configuration from 18,100 to 19,000 pounds. Density altitude ranged from 5000 to 10,000 feet and airspeed from 100 to 186 KCAS. The results of these tests are presented in figures 27 through 32, appendix I.

38. Above 170 KCAS, only small longitudinal control position trim changes were associated with changes in trim speed. Because of the speed stability of the pusher propeller, this characteristic did not adversely affect the pilot's ability to establish a desired trim speed. The lateral control trim position varied linearly from 1 inch left of center, at 100 KCAS to approximately 0.5 inches, left, at 185 KCAS but was not objectionable. There was a slight

left directional trim shift as airspeed was increased from 100 to 185 KCAS. In contrast to the large and objectionable directional control trim shifts which occurred during low-speed flight, directional control trim shifts at high speeds were negligible. The effects of landing gear extension and retraction on the forward flight trim requirements were negligible.

39. Increased collective blade angle produced a nose-up pitching moment which required additional forward longitudinal control. Collective angle changes produced no change in lateral control position. Increased collective required increased left directional control displacement. Changes in gross weight and external configuration produced no significant effect on the forward-flight trim requirements. The forward-flight trim characteristics of the AH-56A are satisfactory.

Trimmability

40. Lateral and longitudinal trim changes are made with a four-direction "coolie hat" switch on the cyclic control. A series of trim corrections was required to trim the lateral and longitudinal forces to zero at all airspeeds including hover. The trim changes in transition from hover to forward flight were adequately compensated for by pretrimming while in hover. Pilot compensation required to trim the aircraft was minimal (HQRS 3) except for exact longitudinal trim at high airspeeds where considerable pilot compensation was required (HQRS 5). This condition is apparently not a function of the trim system but was a manifestation of the weak static longitudinal stability and the long pitch time constant at high airspeeds (above 120 KCAS). The lateral and longitudinal trim systems are satisfactory.

41. Directional trim changes are made with a spring-loaded switch located on the collective pitch lever and operated by the pilot's thumb. Propeller pitch control manipulation is required for a hover transition and the location of the trim switch precludes its use simultaneously with this application. The inability to use directional trim during a hover transition is a shortcoming and should be corrected.

42. The collective pitch control is equipped with an adjustable mechanical friction device and is augmented by a fixed-friction electric brake. The mechanical friction device prevented collective creep in a hover up to the highest blade angle tested (approximately 16 degrees). The electrical friction device was effective in forward flight and the pilot could override the force without difficulty if he chose not to use the spring-loaded disengage lever. The collective control friction devices are satisfactory.

43. A mechanical friction device is provided for the propeller pitch control. At angles greater than 33 degrees, propeller blade angle crept to a lower setting even with the propeller pitch control mechanical friction fully on. Additionally, the propeller blade angle crept during run-up. Slippage of the propeller pitch control is a shortcoming and should be corrected.

Static Longitudinal Stability

44. The collective-fixed static longitudinal stability was evaluated at trim airspeeds from 120 KCAS to 190 KCAS in the external stores configuration with an aft cg at an average density altitude of 4000 feet. During this test, collective and propeller blade angles were held constant at the trim settings. Airspeed was stabilized in 5-knot increments above and below the trim condition by varying longitudinal cyclic control position. Zero sideslip was maintained during these tests. The results of these tests are presented in figures 33 through 35, appendix I.

45. Static longitudinal stability, as evidenced by the variation of longitudinal control position with airspeed, was slightly positive at 120 and 150 KCAS. The gradient about trim at the 120-KCAS point was approximately 0.015 inches/knot and at the 150-KCAS trim point the gradient was approximately 0.012 inches/knots. At airspeeds above the 190-KCAS trim point the gradient was neutral to negative while at airspeeds below the 190-KCAS trim point the gradient was very weak.

46. The static longitudinal control force stability, as evidenced by the variation of longitudinal control force with airspeed, varied greatly over the airspeed range tested. The static longitudinal control force gradient about the 120-KCAS trim point was 0.33 pounds/knot for airspeeds within 5 knots of the trim airspeed. At 5 knots from trim, the longitudinal control force gradient began to shallow and at 20 knots greater than trim and 10 knots less than trim, the gradient became neutral. The longitudinal control force stability was slightly positive about the 150-KCAS trim airspeed. The gradient was approximately 0.01 pounds/knot. About the 190-KCAS trim point, the stability was neutral for points below the trim airspeed and negative for points greater than the trim airspeed. The lateral and directional control trim changes during these tests were minor and not objectionable.

47. Although the static longitudinal stability characteristics of the AH-56A vary from weak to negative, the apparent speed stability is considerably better than that evidenced by the

position and force gradients because of the pusher propeller. The propeller thrust, with propeller blade angle fixed, increases with decreasing airspeed and decreased with increasing airspeed, therefore, improving speed stability. Even though the speed stability is improved, the static longitudinal stability characteristics, especially at high airspeeds, are unsatisfactory.

48. The static longitudinal stability characteristics of the AH-56A do not meet the requirements of paragraphs 3.2.10 and 3.6.3 of MIL-H-8501A, in that the gradient of control position and control force with airspeed is not stable for all flight conditions. The lack of positive static longitudinal stability above 170 KCAS is a shortcoming and should be corrected.

Static Lateral-Directional Stability

49. The static lateral-directional stability characteristics were evaluated at trim airspeeds of 118 to 190 KCAS in the external stores configuration with an aft cg at an average density altitude of 5000 feet. Additionally, qualitative tests were conducted at speeds below 100 KCAS. Sideslips were increased incrementally, left and right, from the zero sideslip trim condition, up to the flight envelope limits. Main rotor collective blade angle, propeller blade angle, airspeed, trim settings, and aircraft ground track were held fixed. The results of these tests are presented in figures 36 through 38, appendix I.

50. Static directional stability, as evidenced by the variation of directional control position with sideslip, was positive about zero sideslip at 190, 150, and 118 KCAS. The gradients were approximately linear to the sideslip limit (app E). Qualitative evaluations below 100 KCAS indicated slightly positive directional stability at sideslip angles up to approximately 25 degrees and essentially neutral at greater sideslip angles. The weak directional stability below 100 KCAS required moderate pilot compensation to adequately control directional attitude and is a shortcoming which should be corrected (HQRS 4).

51. Dihedral effect, as evidenced by the variation of lateral control position with sideslip, was positive about trim for airspeeds above 100 KCAS. The dihedral effect was reduced as airspeed decreased. During the qualitative evaluation below 100 KCAS there appeared to be little change in the lateral stick position as sideslip was increased, indicating neutral dihedral effect. The dihedral effect below 100 KCAS did not meet the requirement of paragraph 3.3.9 of MIL-H-8501A in that it was not positive.

52. Sideforce, as indicated by the variation in bank angle with sideslip, varied from slightly positive at low speeds (below 120 KCAS) to strongly positive at 190 KCAS. Sideforce is the primary cue to sideslip and the lack of adequate sideforce leads to large sideslip excursions. These sideslip excursions cause degradation of turning performance, high vibration levels, and undesirable pitch rates. Maneuvering below 120 KCAS requires moderate pilot effort to maintain control of directional attitude to avoid large sideslip deviations (HQRS 5). The sideforce characteristic of the AH-56A below 120 KCAS is unsatisfactory and is a shortcoming which should be corrected.

53. The static lateral-directional tests revealed a large longitudinal trim shift with sideslip at all airspeeds. Forward longitudinal control displacement and force were required to balance the nose-up pitching moment resulting from right sideslip, and an aft longitudinal control displacement and force were required to balance the nose-down pitching moment resulting from left sideslips. This was not a problem above 120 KCAS; however, with the sideslip excursions which were prevalent below 120 KCAS due to weak directional stability and weak sideforce characteristics, large pitch changes resulted. Extensive pilot effort was required to compensate for these pitch attitude changes (HQRS 6). The excessive pitch due to sideslip is a shortcoming which should be corrected.

54. A deficiency related to static lateral-directional stability characteristics was identified during the APE I testing (ref 1, app I). A similar deficiency is discussed in paragraph 87.

Dynamic Stability

55. Dynamic stability tests were conducted to evaluate the aircraft short-period response following a gust disturbance and long-period response characteristics. Gust disturbances were simulated by making 1-inch pulse control inputs, which were held for 1/2 second. Following the input all controls were held fixed until the aircraft motions damped. Pulse inputs were made at 122 and 152 KCAS at a gross weight of approximately 19,800 pounds and an aft cg, with external stores at a nominal density altitude of 4500 feet. The long-period response characteristics were evaluated by releasing the flight controls to trim during the static stability tests. All stability augmentation systems were engaged during these tests.

56. Representative results of the short-period response tests are presented in figures 39 and 40, appendix I. These data show that the aircraft is well damped in the lateral and longitudinal axes at 122 and 152 KCAS. Qualitatively, aircraft motions following simulated gust disturbances were well damped in all three axes throughout the airspeed envelope. Short-period dynamic stability characteristics of the AH-56A are satisfactory.

57. The long-period response characteristics were qualitatively evaluated at airspeeds up to 190 KCAS. The initial aircraft response, following dynamic releases from off-trim conditions, was toward trim. Because of the coupling between axes, the longitudinal long-period response could not be isolated; however, it appeared to be nonoscillatory and stable. The long-period dynamic stability characteristics of the AH-56A are satisfactory.

Controllability

58. Controllability testing was conducted by stabilizing the aircraft at the test airspeed and making rapid step control inputs (maximum input time 0.2 second) using an adjustable rigid control fixture to control the input size. Tests were performed at airspeeds of approximately 122 and 152 KCAS at a nominal density altitude of 4500 feet. The tests were conducted in the external stores configuration at approximately 19,800 pounds at an aft cg, and the results are presented in figures 41 and 42, appendix I.

59. Longitudinal controllability data indicate a decrease in both control response (maximum angular velocity per inch of control displacement) and sensitivity (maximum angular acceleration per inch of control displacement) with increasing airspeed between 121 and 152 KCAS. The pitch time constant (time between input and development of 63 percent of maximum angular rate) increased slightly with increasing airspeed. These trends with airspeed are probably caused by the pitch densensitizer and contribute to the moderate pilot effort required for precise control of pitch attitude at high airspeeds (HQRS 4). At airspeeds above 170 KCAS, the aircraft response to longitudinal control inputs was noticeably slow to develop. Some roll with lift coupling was noted during these tests but was not objectionable. The long pitch time constant at high airspeeds is a shortcoming which should be corrected.

60. The lateral control response and sensitivity, as well as the roll time constant, are nearly the same at the two airspeeds tested. The maximum roll rate was determined by extrapolating the response data to the limits of lateral control travel. The maximum roll rate was approximately 77 degrees/second (deg/sec) to the right and 42 deg/sec to the left at 122 KCAS, and 77 deg/sec to the right and 49 deg/sec to the left at 152 KCAS. The lateral controllability characteristics were satisfactory at all airspeeds.

61. The sensitivity and response to directional control inputs were qualitatively evaluated throughout the airspeed envelope. High directional control forces resulted in poor harmony between directional and lateral control forces. The directional response and sensitivity caused no problems in achieving desired attitude changes and were satisfactory.

Maneuvering Stability

62. Maneuvering stability was evaluated in left and right steady turns by stabilizing at various bank angles and load factors while maintaining a constant airspeed, constant propeller pitch and constant collective pitch at each airspeed tested. Zero sideslip was maintained to prevent pitching moment contributions due to sideslip. Control positions and forces were recorded at each load factor. Tests were conducted at a nominal density altitude of 4600 feet at airspeeds of 122, 154 and 190 KCAS with collective blade angle settings of 7 degrees, 5 degrees, and 5 degrees, respectively. The aircraft was in the external stores configuration at a nominal gross weight of 20,000 pounds with an aft cg. The data gathered at 122 and 154 KCAS are presented in figures 43 and 44, appendix I. Only qualitative data were obtained at 190 KCAS. Additional qualitative tests were conducted at 80 KCAS, a nominal gross weight of 17,050 pounds, an aft cg, and a nominal density altitude of 7300 feet at 9 degrees collective blade angle.

63. Stick-fixed maneuvering stability, as evidenced by the variation of longitudinal control position with load factor, at 122 and 154 KCAS was positive for very small load factor changes. The stability decreased with increasing normal acceleration and became neutral slightly above 1.3g at 122 KCAS and 1.4g at 154 KCAS. Insufficient quantitative data were obtained at 80 and 190 KCAS to determine the degree of stability. The stick-fixed maneuvering stability of the AH-56A failed to meet the requirements of paragraph 3.3.4 of MIL-F-8785(ASG) which requires that the gradient of the longitudinal control position versus normal acceleration be stable throughout the range of attainable load factors.

64. The stick-free maneuvering stability (longitudinal control force per g) was positive for small increases in load factor and neutral slightly above 1.3g and 1.4g at 122 and 154 KCAS, respectively. The longitudinal control force per g was higher at 154 KCAS than at 122 KCAS, because of airspeed scheduling of the longitudinal force gradient. Insufficient quantitative data were obtained at 80 and 190 KCAS to determine the degree of stability; however, qualitatively a high longitudinal control force was required to reach the envelope limit. The stick-free maneuvering stability of the AH-56A fails to meet the requirements of paragraph 3.3.9 of MIL-F-8785(ASG) which requires that the gradient of the longitudinal control force versus normal acceleration be stable throughout the range of attainable load factors.

65. When the pilot attempted to stabilize at 1.4g at 122 KCAS, the aircraft pitched nose-up and rolled right (fig. 45, app I). The same type excursion occurred when attempting to stabilize at 1.5g at 154 KCAS (fig. 46). The right roll was easily corrected by the appropriate application of lateral control; however, the uncommanded pitch-up resulted in momentary loss of aircraft control (HQRS 10). The uncommanded pitch-up and right roll encountered during these tests are the result of blade moment stall. Uncommanded aircraft motion and loss of control due to blade moment stall was a deficiency during the APE I testing (ref 1, app I). Moment stall has the effect of, suddenly, greatly increasing the aerodynamic nose-down pitching moment of a cambered airfoil. In the case of the AH-56A, the increased blade pitching moment feeds back to the control gyro which, in turn, causes aircraft response. Blade moment stall is a function of blade angle of attack and therefore the following two methods may be used to avoid the stall: (1) maneuvering at low load factors, and (2) lowering main rotor collective blade angle while maneuvering. The first method degrades the aircraft load factor capability to about half the present envelope limits and greatly increases turning radii and return-to-target times. The second method degrades aircraft performance. A light airframe buffet and a moderate increase in vibrations preceded blade moment stall; however, these cues are insufficient to prevent the aircraft from becoming uncontrollable within the allowable flight envelope during normal maneuvering. No uncommanded aircraft motions were encountered at envelope limit load factors of 1.43 at 190 KCAS and 1.5 at 80 KCAS. Blade moment stall causes loss of aircraft control within the flight envelope which precludes safe accomplishment of the attack helicopter mission, and is a deficiency which must be corrected.

Table 8. Simulated Engine Failure Characteristics.

Entry Airspeed (KCAS)	Flight[1] Condition	Entry Engine Power (shp)	Entry Collective Blade Angle (deg)	Auto. delta -beta Delay (sec)	Manual beta Control Delay (sec)	Collective Control Delay (sec)	Minimum Transient Rotor Speed and Time (%)	(sec)	beta angle after delta-beta (deg)	beta change due to delta-beta (deg)
59	level	1738	9.1	N/A[2]	1.30	1.00	86	2.6	N/A[2]	N/A[2]
63	level	1785	10.1	N/A[2]	1.88	2.00	82	2.9	N/A[2]	N/A[2]
122	level	1938	6.9	1.13	2.20	2.30	85	2.5	24.5	2.1
120	900 fpm climb	3202	7.1	0.82	0.82	1.13	86	1.6	N/A[3]	N/A[3]
123	770 fpm climb	3481	6.8	0.81	27.00	2.17	68	8.6	24.9	11.2
154	level	2860	5.1	0.96	1.73	2.00	87	1.6	24.2	9.1
167	900 fpm dive	2765	4.9	1.00	1.85	2.84	87	2.6	24.6	9.5
188	1810 fpm dive	3140	4.6	0.89	1.06	2.00	90	1.1	data unreliable	

[1] Entry rotor speed: 100 percent.
Average density altitude: 2700 feet
Average gross weight: 20,130 pounds.
Center-of-gravity: aft (FS 299.8)

[2] Entry beta setting below delta-beta actuation setting.

[3] Manual beta and delta-beta activated simultaneously.

Simulated Engine Failure Characteristics

66. Instantaneous engine failures were simulated by use of the engine overspeed protection system which restricts fuel flow. All flight controls were held fixed as long as practical and propeller pitch (beta) control and collective control inputs delayed until reaching a target minimum rotor speed. An automatic propeller pitch control system (delta beta) reduces propeller pitch in case of engine failure at high propeller blade angles. The test conditions and results are presented in table 8. At initial propeller pitch settings greater than +18 degrees, the delta-beta system is designed to reduce propeller pitch angle to approximately +18 degrees and the pilot must further reduce the angle to +8 degrees, the autorotational setting.

67. During this test, the delta-beta brought the propeller pitch angle to approximately +24 degrees which reduced the rate of decay of main rotor speed considerably. Failure to further reduce propeller pitch manually, resulted in a very low rotor speed (68 percent) on the 123-KCAS climb entry. Although the rotor speed dropped below the minimum transient rotor speed of 85 percent on this entry, aircraft control was not lost and recorded structural loads were not excessive. A subsequent aircraft inspection showed no damage. The minimum transient rotor speed was also exceeded by 3 percent on the 63-KCAS level-flight entry following a 2-second delay of collective control input. This delay time failed to meet the requirement of paragraph 3.5.5 of MIL-H-8501A, in that within 2 seconds of engine failure the main rotor speed fell below the safe minimum rotor speed, as defined by the safety-of-flight release. Previous testing (ref 1, app A) indicated that the rapid rate of rotor speed decay was a deficiency. A more extensive investigation during these tests showed that it was not a problem area.

68. The initial cues to the pilot at all airspeeds, left yaw and the noise reduction associated with decreasing rotor speed, were adequate. A slight right roll followed the left yaw. Both yaw and roll excursions were easily and naturally corrected by the pilot. A slight pitch-down was experienced at the high airspeeds but it was easily corrected and controlled. Desired airspeed control was not difficult to maintain but the general lack of airspeed cues required the pilot to rely on the airspeed indicator. At slow entry airspeeds (60 KCAS), the collective pitch control was critical because of high collective blade angles and low propeller pitch angles. At intermediate entry airspeeds (120 KCAS), both propeller

pitch and collective pitch controls were critical. At higher airspeeds (188 KCAS) the propeller pitch control was the most critical. Following an engine failure, the pilot must decide which of the controls are critical in addition to the normal pilot effort required for autorotational entries. This moderate pilot effort required to concurrently manipulate two critical controls to prevent low rotor speed during autorotational entries is a shortcoming which should be corrected (HQRS 4).

Autorotational Characteristics

69. A qualitative evaluation of the handling qualities in steady-state autorotational flight was conducted at approximately 95 KCAS, the contractor recommended autorotational airspeed. Attitude and airspeed control were satisfactory. The rate of descent was approximately 2200 feet per minute at an average gross weight of 19,700 pounds, an aft cg, and an average density altitude of 2700 feet. Left and right turns were made at approximately 30-degree bank angles with minimal pilot workload (HQRS 3). The rotor speed increased in a right turn and decreased in a left turn, but rotor speed, airspeed, and attitude control were no problem. Autorotational landings were prohibited by the safety-of-flight release; however, minimum power descents to a run-on landing were accomplished without difficulty. Further testing would be required to completely evaluate autorotational landings.

Automatic Stabilization System Characteristics

70. Simulated failures of two of the automatic stabilization devices, the roll compensator and the pitch desensitizer, were conducted at 2000-, 6000-, and 8000-foot density altitudes in the clean and external stores configurations.

71. The roll compensator, which operates from hover to 80 KCAS, was failed OFF at a hover, in transition to forward flight, and in transition to a hover by pushing the roll compensator button to deactivate the system. Hover takeoffs and landings were also made with the roll compensator OFF throughout the maneuver. No change in handling qualities were noted and there was no noticeable aircraft reaction when the system was deactivated. The recorded structural loads showed no increase during the simulated failures. The roll compensator failure characteristics are satisfactory.

72. The pitch desensitizer, which operates above 110 KCAS, was failed OFF in maneuvering flight at load factors of 1.3g in level turns and in nap-of-the-earth flight at airspeeds of 110, 145, and 170 KCAS by pushing the pitch desensitizer button. At 145 KCAS and below, no noticeable difference could be seen and the aircraft exhibited no response to the simulated failures. At 170 KCAS the sensitivity of longitudinal control inputs was increased but was not uncomfortable and caused no noticeable deterioration of mission capability. Coupling between pitch and roll was also noticeable with rapid lateral control inputs but was acceptable. The pitch desensitizer failure characteristics are satisfactory.

MISCELLANEOUS ENGINEERING TESTS

Cockpit Evaluation

73. The cockpit of aircraft S/N 66-8834 (ship 1009) was in an engineering flight test configuration. Many mission-essential items were replaced by special test instrumentation. The results of this evaluation are as follows:

a. The propeller pitch control and the engine condition control are similar (rotating grips) and are both located on the collective pitch lever with the propeller pitch control immediately forward of the engine condition control. The similarity could cause serious problems if the engine rpm was reduced when reduced propeller pitch was desired, or in the case of tail rotor failure in a hover, if propeller pitch was reduced when engine idle was desired. The similarity in configuration and close proximity of the engine condition and propeller pitch controls is a shortcoming.

b. There was no means of shutting OFF the engine from the cockpit in case of total electrical failure, which is a shortcoming.

c. The location of the ALTERNATE N_f increase/decrease switch is such that the pilot must remove his hand from a primary flight control to use it, which is a shortcoming.

d. The nonessential circuit breaker panel is not accessible to either crewmember in flight, which is a shortcomming.

e. The readability of rectangular gages for engine torque, turbine inlet temperature, gas producer speed (N_G), free turbine speed (N_f), and rotor speed (N_R) was unsatisfactory because of small size and poor sensitivity. This is a shortcoming. The test instrumentation turbine inlet temperature circular indicator incorporates a digital readout window on the dial face and was easy to read.

f. The fault locator aural warning system (FLAWS) and the voice warning system are excellent and should be included in future designs.

74. The environmental control unit (ECU) was inadequate to provide crew comfort even though the equipment cooling line was closed off. The inadequate ECU is a shortcoming.

75. The field of view from the rear cockpit to the front of the aircraft was restricted by the instrument panel during takeoffs, landings, and low-level mission maneuvers. This restriction was most apparent during low-level mission maneuvers (50 to 100 feet). The inadequate forward field of view from the aft cockpit is a shortcoming and should be corrected.

Weight and Balance

76. The weight and balance of the test aircraft was determined prior to the start of testing. All test instrumentation was installed prior to this test and the aircraft was weighed with no fuel and without the external stores and pylons installed. The weight and cg obtained for the empty aircraft (test instrumentation installed) were 14,632 pounds and FS 313.2, respectively.

Ground Operation Characteristics

77. Access to the crew compartment is via a walkway over the right sponson leading to a folding walkway under the cabin access doors. The walkway is adequate; however, it is narrow, requiring extra care by crewmembers and maintenance personnel. Adequate handholds are not provided along the walkway, which is a safety hazard. The lack of adequate handholds is a shortcoming and should be corrected.

78. The starting and preflight run-up procedures are very simple and straightforward. The auxiliary power unit (APU) is started by holding a spring-loaded start switch momentarily in the start position and then releasing it. The main engine is started by momentarily depressing a start button and rotating the engine condition grip to the IDLE position at 20-percent gas producer speed. The remainder of the start cycle is automatic. The start may be aborted by momentarily depressing an abort-stop button. Starts may be accomplished with the rotor brake on; however, the brake must be released for engine settings above ground idle. The rotor brake, which is an excellent feature, is very effective in stopping the rotor and may be used after engine shutdown and after the main rotor speed is at or below 40 percent. The engine run-up and aircraft systems checks are simple.

79. Ground taxi is accomplished by use of propeller thrust for acceleration, tail rotor thrust for directional control, and reverse propeller thrust and/or wheel brakes for deceleration and stopping. The wheel brakes are satisfactory and all other aspects of ground taxi are excellent. The tail wheel is required to be locked for all takeoffs and landings as well as for high-speed taxi (above 20 knots). The tail wheel must be unlocked for all taxi turns since directional control is by tail rotor thrust. It is possible to pull the tail wheel unlocking lever in the cockpit, by overriding the spring tension, without actually unlocking the tail wheel. This occurred when there was a side load on the tail wheel. Subsequent turns, which maintain the side load, will cause the tire to skid; however, pedal input in the opposite direction will unlock the tail wheel. Though annoying, this characteristic is advantageous since it prevents the pilot from unlocking the tail wheel with a side load which would turn the aircraft. The pilot can check to insure the tail wheel is unlocked by applying small left and right direction control inputs. The ground taxi characteristics are satisfactory.

Engine Characteristics

80. Guaranteed power-available is presented in figures 47 through 49, appendix I. The data presented in these figures have been corrected for installacion losses including inlet temperature rise and pressure loss (figures 50 and 51). Power-available data are presented for maximum rated power (10 minute limit), military rated power (30 minute limit), and normal rated power (maximum continuous power). Test data obtained during hover performance tests are compared to guaranteed engine performance data in figure 52). These data indicate that the test engine met the guaranteed engine performance.

81. Power management required higher pilot workload in this aircraft than in conventional, turbine-powered helicopters because of the additional control requirement of the pusher propeller. Changes in propeller pitch must be carefully programmed with collective pitch during transition to forward flight to avoid exceeding torque and turbine inlet temperature limits. In a hover takeoff the propeller pitch was increased to maintain the desired acceleration schedule as collective pitch was decreased. The poor readability and sensitivity of the torque indicator and the nonlinearity of the propeller pitch control contribute to the power management workload. It was very difficult to maintain a constant torque setting because of the dual power controls. The torque and turbine inlet temperature limits were exceeded on

several occasions during hover takeoff. A similar power management problem existed at high power settings in forward flight. Relatively large changes in power accompanied small changes in propeller pitch at angles above approximately +28 degrees. The pilot workload increased significantly during maximum power operations, which required undue pilot attention to prevent exceeding the engine limits. This high power management workload is a shortcoming and should be corrected.

Airspeed System Calibration

82. Airspeed data were corrected for position error using a Lockheed-provided calibration (fig. 53, app I) of the test (boom) airspeed system. The standard ship's system was not calibrated during these tests.

Vibration Characteristics

83. No quantitative data were obtained on cockpit vibration levels during this evaluation because the contract did not provide for installation of the necessary sensors. The vibration levels were determined to be a deficiency in APE I testing (ref 1, app I); however, APE I was flown in a different configuration and with vibration instrumentation installed. The vibration levels discussed herein, are qualitatively presented to give an indication of those regimes where the greatest vibration levels occurred. Vibration levels were uncomfortable during transition from hover to forward flight, in transition from forward flight to hover, and in forward flight at collective settings greater than 5 degrees. The vibrations in transition were highest in the external stores configuration; however, they were of short duration. The highest vibration level occurred at 115 KIAS in the external stores configuration with four empty 2.75-inch rocket pods and collective settings above 5 degrees. The uncomfortable vibration level in the areas stated is a shortcoming which should be corrected.

MISSION SUITABILITY TESTS

Qualitative Evaluation of Typical Mission Maneuvers

84. The mission maneuver capability of the AH-56A was evaluated during low-speed flight, bob-up (vertical climb) and lateral evasive flight from a hover, high-speed nap-of-the-earth flight, and target tracking tasks in a dive. The evaluation was conducted during day, visual flight conditions at density altitudes to 5000 feet in the external stores configuration at gross weights from 18,300 to 19,700 pounds.

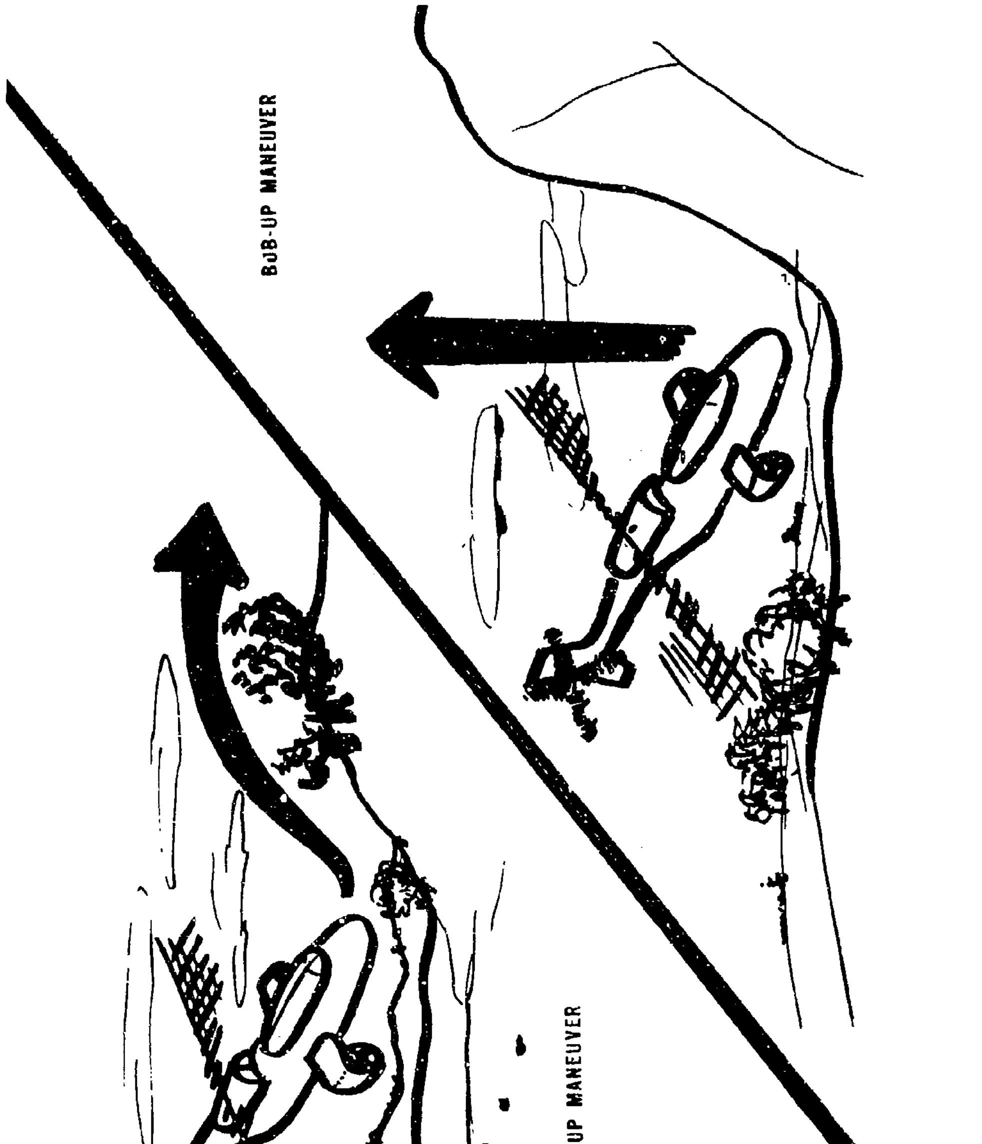


FIGURE C. POP-UP AND BOB-UP MANEUVERS.

85. Forward accelerations from hover to 60 KCAS and decelerations from 60 KCAS to hover were qualitatively evaluated. Large control trim shifts and vibrations occurred. Moderate pilot effort was required to accomplish these maneuvers; however, as the available power margin decreased, the pilot effort required for power management increased (HQRS 4). During flight over rolling terrain at less than 50 feet and at airspeeds from 40 to 80 KCAS, control response was satisfactory and no large control trim shifts were observed. Large pedal forces were required to rapidly turn the helicopter and the cyclic and pedal force were not in harmony. Under good visibility conditions, minimal pilot effort was required to maneuver the helicopter at slow speed close to the ground (HQRS 3) when the power margin was adequate.

86. The pop-up and bob-up maneuvers are illustrated in figure C. Pop-up maneuvers were accomplished from 40 KCAS in low-level flight keeping propeller pitch constant. A climb was made over masking terrain and target acquisition was simulated while remaining at low level. Break-off and reversal at 75 KCAS required minimal pilot effort (HQRS 3). The handling characteristics during the pop-up maneuver were satisfactory under good visibility conditions. A bob-up from a hover was accomplished to evaluate handling characteristics during simulated mask breaking and target acquisition. No control problems were observed except that with a limited power margin, moderate pilot effort was required to monitor the engine limits (HQRS 4).

87. Directional control was extremely difficult below 120 KCAS when long range external visual cues were not used. This difficulty was evidenced by frequent attitude disorientation and an inability to maintain coordinated flight, resulting from a combination of poor static lateral-directional stability characteristics (paras 50 through 53), excessive breakout forces and force gradients of the directional control system (para 19), and poor harmonization of the directional and lateral/longitudinal control forces (para 19). Attack helicopters must be able to accomplish the slow-speed, low-level mission, to include navigation, target acquisition, target tracking, and evasive action, in adverse weather. Restricted visibility and lack of a definite horizon are realistic battlefield conditions. Pilot attention would be totally devoted to controlling the aircraft under these conditions and adequate performance could not be attained with maximum tolerable pilot compensation (HQRS 7). The inability to effectively perform slow-speed, low-level mission tasks below 120 KCAS under adverse weather conditions, due to the combined effect of the lateral-directional control system and static lateral-directional stability characteristics, is a deficiency which must be corrected.

88. High-speed flight was conducted over rolling terrain at less than 100 feet and at airspeeds between 120 and 180 KCAS. No unusual control motions were necessary and control response was good during flight requiring small changes of attitude. Moderate pilot effort was required for power management since small changes of propeller pitch, at high blade angles, caused relatively large changes in power (HQRS 4). The ability to maneuver within the load factor envelope was limited by blade moment stall at airspeeds from 120 to 150 KCAS. This limitation was especially true in light-to-moderate turbulence. Blade moment stall causes loss of aircraft control which precludes the continuation of any mission task (HQRS 10).

89. Target acquisition, tracking, and target shifting tasks were evaluated by rolling into a simulated firing pass from airspeeds of 100 and 150 KCAS. Propeller pitch was adjusted to control airspeed. Dives were performed at various combinations of dive angle, airspeed, and propeller pitch settings, including accelerating, decelerating, and stabilized airspeed dives. The ability to select a wide range of airspeeds independently of dive angle is an extremely valuable characteristic that enhances the capability of the AH-56A to deliver fire on a target. Initial target acquisition was easy with minimal time required. Target tracking during the descent was satisfactory with minimal pilot effort require to track and maintain airspeed (HQRS 3). Only slight pitch and yaw oscillations were observed. Sideslips developed during rapid target shifts causing noticeable roll, and yaw oscillations. These oscillations varied in magnitude depending on the degree of sideslip. Moderate pilot effort required to rapidly shift targets during a dive is a shortcoming which should be corrected (HQRS 4).

Foward Area Concealment

90. A limited forward area concealment evaluation was conducted using manpower, 1/4-ton, and 3/4-ton tactical military trucks to move the aircraft over various desert surfaces. Compatibility of the tow bar hook-up with a 2-1/2 ton tactical truck was also verified. The terrain surface was measured with a cone penetrometer and the average airfield index (AI) was determined. The aircraft was at a gross weight of 16,729 pounds and an aft cg (FS 302.6) with the main rotor blades removed.

91. The lowest portion of the fuselage was 21 inches from the ground. This low point was the belly turret located at fuselage station 190. The primary aircraft design aspect that interferes with concealment operations is the main rotor blades. The blades cannot be folded and would have to be removed in order to move the aircraft into a tree

line. Removal of the blades is not practical for concealment purposes. The full-swiveling tail wheel allowed the aircraft to be turned using a main landing gear as a pivot point. The turning radii are as follows:

Landing gear - 5 feet 2 inches
Outer wing tip - 12 feet 10 inches
End of the horizontally extended antitoruqe blade -- 34 feet 6 inches

92. Six men were required to manhandle the aircraft on a smooth level ramp. To move the aircraft forward required two men on each side of the aircraft pushing on the wing roots and/or wing trailing edges, one man guiding the aircraft with the tow bar, and a safety man in the cockpit. To move the aircraft aft, the crewmembers pushed on the leading edge of the wings and wing root. The handhold hardpoints or pushing points, and the no-push vulnerable areas (antennas, etc) were not marked. The push points were adequate for ground handling the aircraft. Ten men were required to move the aircraft over an improved hard area (AI 10+) with a rough and rocky surface. Moving the aircraft by manpower in soft terrain was impractical.

93. The aircraft was moved over various desert terrain surfaces including firm and relatively smooth, sun-baked desert floor and rough and soft surfaces, A strip of ground 75 feet by 125 feet was plowed to a depth of approximately 36 inches and then smoothed with a grader. The average AI for this area varied from 4 to 6. This equates to a California Bearing Ratio (CBR) of 2.4 and 4.2. The plowed soil consisted of various dirt and rock distribution with the highest percentage being a very fine powered dirt. The rock distribution was small in comparison, and ranged from a pea size to about the size of a quarter. The aircraft can be prepared for towing by three men in approximately 25 seconds. The tow bar was connected to the tail wheel and the aircraft was pulled across the test surface. There were no problems initially using the 1/4 ton truck; however, at the soft end (CBR 2.4) the prime mover became stuck. The primary cause was the dirt piling up ahead of the small tail wheel. The small tail wheel (5.00 x 4), is high pressure (100 psi) and thus provides a minimum of flotation; however, the towbar acts as a skid and provides bonus flotation. The 3/4-ton truck moved the aircraft through on the first pass in 45 seconds. On the second pass the truck became momentarily mired but was able to continue after some maneuvering. The aircraft main wheels made ruts 4 to 6 inches deep and the tail wheel dug in approximately 10 inches. The prime mover ruts were from 6 to 10 inches deep. Since the 2 1/2-ton truck has better flotation than the other vehicles utilized, there was no doubt of its capability to tow the aircraft under the same conditions. The universal tow bar was compatible with the 2 1/2-ton truck. The higher positioning of the pintle on the 2 1/2-ton truck should reduce the digging in of the tail wheel.

Maintenance Characteristics

94. A maintenance evaluation was performed throughout the flight test program by a three man maintenance team. All failures and maintenance actions performed during the test program were recorded. The small number of flight hours provided limited opportunity to observe component repair/replacements, thus necessitating a qualitative evaluation of the aircraft in lieu of the desired quantitative evaluation. No formal remove/replace tests were allowed, and the team was instructed to perform the evaluation on a noninterference basis. The aircraft was fully instrumented, which resulted in maintenance complications that would not normally exist on operational aircraft. The climatic environment and maintenance facilities were more favorable than would be experienced under combat conditions. These factors tend to minimize the maintenance tasks and maintenance task times. Maintenance evaluations are given for these categories: (1) Airframe/Landing Gear/Fuel System, (2) Engine/APU, (3) Flight Controls/Main Rotor/Transmission/Drive Train, (4) Hydraulics, and (5) Instruments/Cockpit/Electrical.

95. The Airframe/Landing Gear/Fuel System category was observed to have maintenance problems. Convenient work platforms, foot and handholds are provided except for the tail rotor area and the left side of the crew stations. Work platforms should be provided for these two areas. The aircraft has intricate mechanical linkage between the engine cowl release levers on either side of the aircraft to permit single cowl latch release. Two men are required to open the engine cowl due to the size and binding of the cowl on the track. The binding should be eliminated, or the linkage between the latches be removed. Integral support arms, for support of the transmission cowl when open, should be provided. Aircraft drainage points are provided, but are of poor design, an example of which is the environmental control unit cowling drainage point. Inadequate drainage will induce corrosion and drastically increase field maintenance time. The tail landing gear strut was replaced twice during the test due to leaking of hydraulic fluid at the lower seal. Elimination of the tail wheel retraction capability would reduce cost and maintenance problems on the strut and hydraulic plumbing. The maintainability qualities of the Airframe/Landing Gear/Fuel System category are better than have been seen in past fielded Army aircraft. However, attempts to better these qualities through the development cycle have, in some cases, resulted in complexity of design and induced added maintenance.

96. The Engine/APU category had several excellent design features, but also several shortcomings. The engine cowling slides back to permit easy access to the area between the forward mounting flange and the exhaust cowling. The use of band fasteners on many engine components facilitates rapid removal of these components. However, several high maintenance components should be relocated to improve accessibility. An example of this is the oil filter which is located beneath the midpoint of the engine. Significant improvements should be made in the routing of line assembiles and wire bundles. Another problem is the rapid build-up of exhaust emissions under the exhaust cowling and on the aft fuselage. The cleaning required will result in a significant increase of maintenance time at organizational level. These emissions can induce corrosion problems and cause early deterioration of the exterior paint. The maintainability characteristics of the Engine/APU category are an improvement over existing fielded Army aircraft.

97. The Flight Controls/Main Rotor/Transmission/Drive Train category posed several maintenance problems. Although the servo package utilizes the modular concept, it has introduced maintenance problems resulting from its weight of 342 pounds and cumbersome size. Accessibility is limited by a relatively small access opening as compared to the servo package. Complexity of this unit may require a high degree of skill for removal and replacement/rigging. The simplicity of the rotor system results in minor maintenance tasks being required for normal operation. Sight gages, which are an excellent design feature, are used on components requiring periodic servicing. The Flight Controls/Main Rotor/Transmission/Drive Train components are very accessible, except for the servo package.

98. Several maintenance problems were encountered in the hydraulic system. The accumulator pressure gages were not visible. It is recommended that these gages be made readable without the use of a mirror. Unnecessary complexity was found in the design of the unique fluid replenishment system. A less complex replenishment system should be used since maintenance will be improved, and a significant cost reduction will be realized.

99. The Instruments/Cockpit/Electrical category was observed to have few maintenance problems. However, one area that needs improvement is the accessibility of the battery compartment. It is desirable that the battery be easily serviced and removed from a single access opening. The accessibility of the battery should be improved and all maintenance work on the battery be performed through a single access opening.

100. The number of GSE and Special Tools required to maintain this aircraft have been held to a minimum. The basic Army mechanic's tools are sufficient for many routine maintenance tasks. Fifty-four percent of the GSE/Special Tools are presently available in the DSA.

101. Except for the maintenance shortcomings described in this section, this aircraft has good maintainability qualities, including adequate accessibility and excellent provisions for ease of performing scheduled maintenance. The maintenance shortcomings should be corrected.

CONCLUSIONS

General

102. The following conclusions were reached upon completion of the attack helicopter evaluation of the AH-56A Cheyenne compound helicopter.

a. Forward flight acceleration performance to 120 KCAS was excellent (para 15).

b. Forward flight deceleration performance was outstanding at all airspeeds (para 15).

c. The capability of using propeller pitch to vary pitch attitude in a hover and during slope landings is an enhancing feature not normally found in pure helicopters (para 24).

d. The fault locator aural warning system (FLAWS) and the voice warning system were excellent features (para 73f).

e. The ground taxi characteristics using the pusher propeller were excellent (para 79).

f. The ability to select a wide range of airspeeds independently of dive angle enhances mission capability (para 89).

g. Two deficiencies and 24 shortcomings were identified.

Deficiencies and Shortcomings Affecting Mission Accomplishment

103. Correction of the following deficiencies is mandatory:

a. Loss of aircraft control within the flight envelope resulting from blade moment stall (HQRS 10) (para 65).

b. The inability to effectively perform slow-speed, low-level mission tasks below 120 KCAS under adverse weather conditions, due to the combined effect of the lateral-directional control system and static lateral-directional stability characteristics (HQRS 7) (para 87).

104. Correction of the following shortcomings is desirable. These shortcomings are listed in the order that they appear in the text and not necessarily in the order of importance.

a. Excessive breakout force in the directional control pedals (para 19).

b. Excessive directional control force gradients (para 19).

c. High pilot workload during transition from hover to forward flight (HQRS 6) (para 25).

d. Excessive pilot workload required to maintain desired heading during sideward acceleration at and above 20 KTAS (HQRS 6) (para 34).

e. Excessive pilot effort required to control aircraft attitude and maintain altitude in a rapid reversal of lateral acceleration (HQRS 6) (para 35).

f. Inability to use directional trim during a hover transition (para 41).

g. Slippage of the propeller pitch control (para 43).

h. Lack of positive static longitudinal stability above 170 KCAS (para 48).

i. Weak directional stability below 100 KCAS (HQRS 4) (para 50).

j. Lack of adequate sideforce at airspeeds below 120 KCAS (HQRS 5) (para 52).

k. Excessive pitch due to sideslip (HQRS 6) (para 53).

l. Long pitch time constant at high airspeeds (HQRS 4) (para 59).

m. Moderate pilot effort required to concurrently manipulate two critical controls to prevent low rotor speed during autorotational entries (HQRS 4) (para 68).

n. Similarity in configuration and close proximity of the engine condition and propeller pitch controls (para 73a).

o. Lack of means of shutting off the engine from the cockpit in case of total electrical failure (para 73b).

p. Unsatisfactory location of the ALTERNATE N_f increase/decrease switch (para 73c).

q. Inaccessability of the nonessential circuit breaker panel in flight by either crewmember (para 73d).

r. Poor readability of the rectangular gages for certain engine parameters (para 73e).

s. Inadequate environmental control unit cooling (para 74).

t. Inadequate field of view from the aft cockpit (para 75).

u. Lack of adequate handholds near the access walkway (para 77).

v. High power management workload (para 81).

w. Uncomfortable vibration level in portions of the flight envelope (para 83).

x. Moderate pilot effort required for rapid target shifts (HQRS 4) (para 89).

Specification Compliance

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

a. Paragraph 3.2.4 - The longitudinal control force gradient exceeded the 2.0-lb/in. limit of this paragraph (para 19).

b. Paragraph 3.3.13 - The directional breakout force to the right exceeded the 7.0-pound limit specified and the maximum directional control force exceeded the 15-pound limit (para 19).

c. Paragraphs 3.2.10 and 3.6.3 - The longitudinal control position and control force stability with respect to airspeed were not stable about the 190 KCAS trim airspeed (para 48).

d. Paragraph 3.3.9 - The variations of lateral control displacement with steady sideslip angle were not stable at all the speeds specified (para 51).

e. Paragraph 3.5.5 - Following a simulated engine failure at 63 KCAS, the rotor speed fell below the minimum safe transient rotor speed following a 2-second delay on collective control input (para 68).

106. 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 - The aircraft has neutral stick-fixed maneuvering stability (para 63).

b. Paragraph 3.3.9 - The aircraft has neutral stick-free maneuvering stability (para 64).

RECOMMENDATIONS

107. The deficiencies identified during this evaluation must be corrected (para 103).

108. The shortcomings, correction of which is desirable, should be corrected (para 104).

APPENDIX A. REFERENCES

1. Final Report, USAASTA, Project No. 70-02 and 71-17, *Army Preliminary Evaluation I and Research and Development Acceptance Test I, AH-56A Cheyenne Compound Helicopter*, March 1972.

2. Letter, AVSCOM, AMSAV-EFT, subject: Test Request, Attack Helicopter Evaluation of the AH-56A Helicopter, Project No. 72-08, 9 March 1972.

3. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements for*, 7 September 1961, with Amendment 1, 3 April 1962.

4. Military Specification, MIL-F-8785(ASG), *Flying Qualities of Piloted Airplanes*, 1 September 1954.

5. Preliminary Operational/Maintenance Manual, POMM 15-1520-222-10, *Operator's Manual, Helicopter, Attack, AH-56A (Lockheed)*, July 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-716(ST) engine rated at 4330 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. The engineering 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. 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. 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 engineering test aircraft is not configured with the weapons systems.

5. A detailed description of the flight control system is contained in appendix C.

MAIN ROTOR

6. 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.

7. 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.

8. 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.

9. 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_{m_0} and C_{m_a} 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. 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	3 deg, 10 min
Sweep, forward	4 deg, 00 min
Disc area, πR^2	2062 ft2
Blade area, bcR	239.1 ft^2
Solidity, σ = bc/πR	0.1159
Geometric twist, $\theta 1$, 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, $\theta 0$	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 on inertia includes main rotor, tail rotor, and propeller	10,742 slug-ft^2

TAIL ROTOR

10. 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	1019380
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	-7.4 deg to 24.2 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 on inertia	12.6 slug-ft^2

PROPELLER

11. 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.

12. 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 is restricted to -12 degrees.

13. An automatic system (delta beta) senses main rotor shaft torque and load factor to provide a reduction of propeller pitch to approximately +18 degrees or from reverse pitch to approximately -7 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 66-8834	-12 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

14. 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. The resulting section defies aerodynamic description. Compensation for rolling moment due to propeller torque is provided by an increased incidence angle on the right wing. Detuning weights were installed in 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	1 deg, down
Twist:	
Left wing	-3 deg, 6 min
Right wing	-3 deg, 2 min

HORIZONTAL STABILIZER

15. 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. 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

16. 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. FLIGHT CONTROL DESCRIPTION

1. 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. 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. 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.

2. The pitch control system includes four augmentation devices designed 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 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 augmentattion device was designed to reduce pitch-due-toroll cross-coupling. This system applies longitudinal control inputs through the desensitizer moducation 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).

3. 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 $\pm$0.329 inch of lateral control displacement. Two notch filters are provided to suppress 16-hertz and 32-hertz vibratory inputs to the roll SAS. 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.

4. 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	$\pm$15 deg

Stick throw:	
Longitudinal	11.0 in (approx)
Lateral	7.5 in.
Control input rotation	36.0 deg (advanced)
Gyro moment per inch of stick:	
Longitudinal	278 ft-lb/in.
Lateral	337 (45%) ft-lb/in.
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) 0.28 lb/in./sec(225 kt)
Lateral	0.115 lb/in./sec(hover) 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	6.75 in. total
Trim authority:	
Left pedal	100 percent
Right pedal	90 percent
Servo rate limits (no load)	3.75 in./sec

APPENDIX D. PHOTOGRAPHS

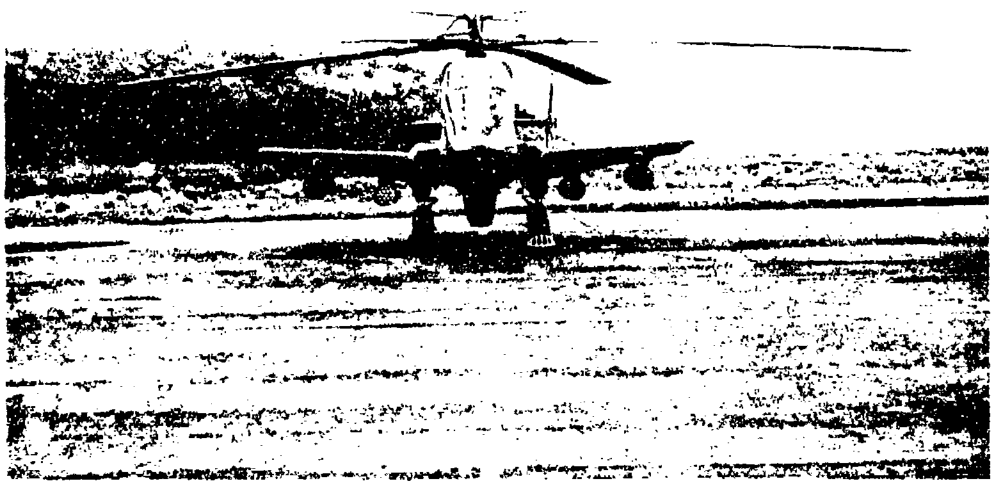


PHOTO 1. FRONT VIEW

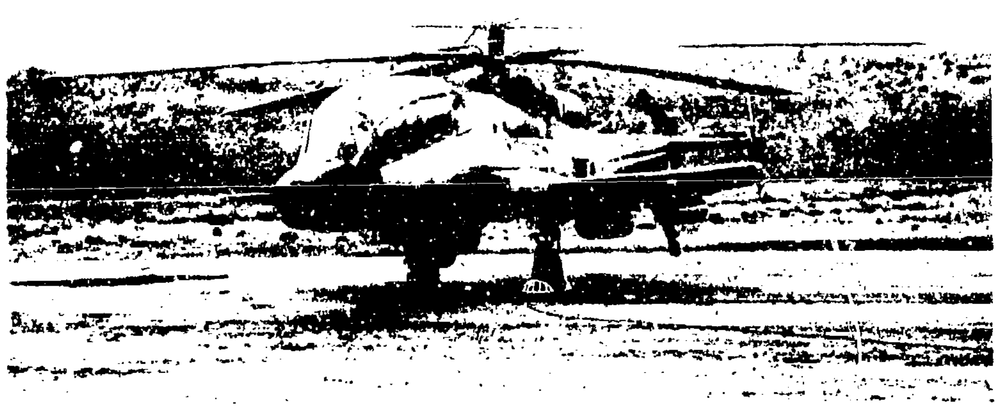


PHOTO 2. FRONT QUARTERING VIEW

PHOTO 3. SIDE VIEW

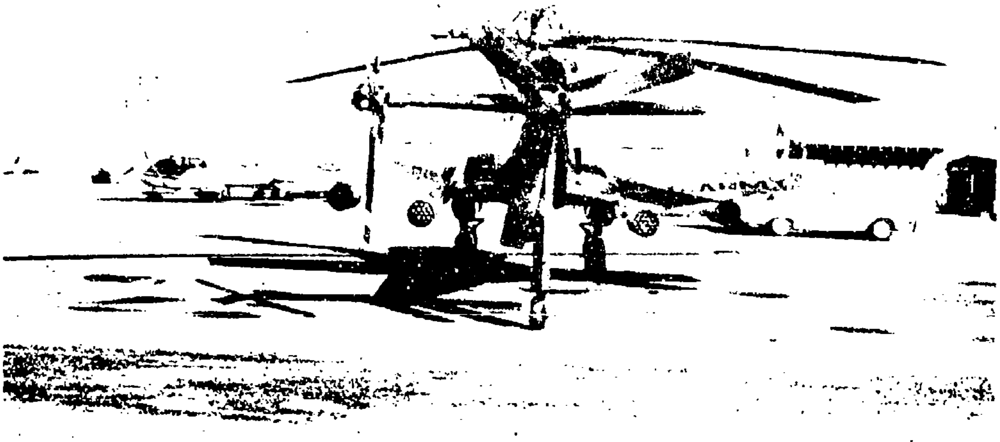


PHOTO 4. REAR VIEW

PHOTO 5. SLOPE LANDINGS

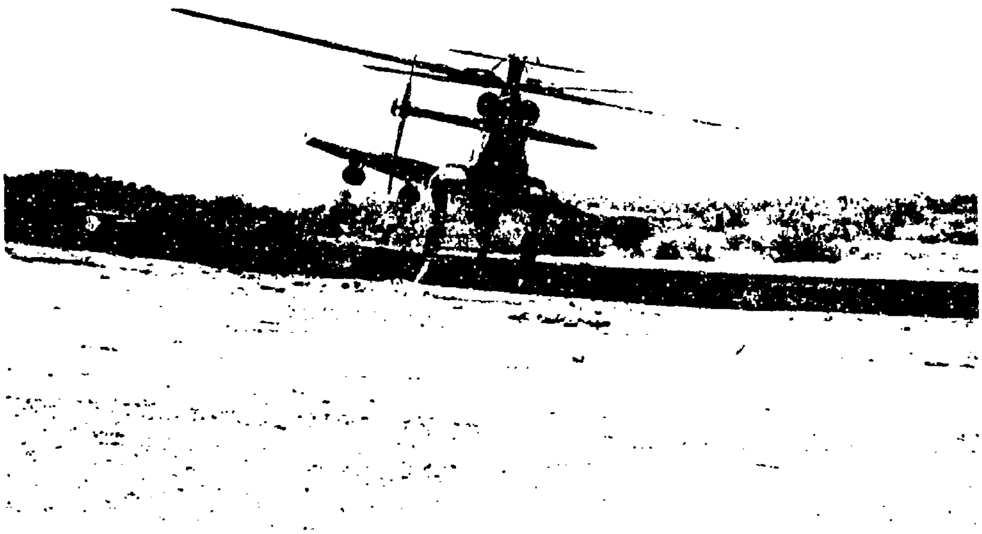


PHOTO 6. SLOPE LANDINGS

APPENDIX E. SAFETY OF FLIGHT RELEASE

This appendix contains the safety-of-flight release, amendments, and flight envelope for the Attack Helicopter Evaluation of the AH-56A Cheyenne Compound Helicopter.

DEPARTMENT OF THE ARMY

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

6 June 1972

AMSAV-EFT

SUBJECT: Revision of Safety of Flight Release for AH-56A Attack Helicopter Evaluation Dated 15 May 1972

Commanding Officer
US Army Aviation Systems
Test Activity
ATTN: SAVTE-P

1. Reference is made to Safety of Flight Release for AH-56A Attack Helicopter Evaluation Dated 15 May 1972.

2. The purpose of this letter is to amend the Ref 1 Safety of Flight Release to permit sideward and rearward flight at higher altitudes. Specifically para 4a (5) is amended to permit sideward flight to 35 KTAS at w/σ (gross weight/density ratio) up to 21,000 lbs, decreasing linearly to 15 KTAS at a w/σ of 24,000 lbs. ~~Rearward flight is permitted to 30 KTAS at w/σ up to 24,000 lbs.~~ During rearward flight increases in reverse propeller thrust will be required to minimize buffet of empennage surfaces as rearward speed is increased.

REARWARD FLT UP TO 30 KTAS

FOR THE COMMANDER:

CHARLES C. CRAWFORD
Chief, Flt Std & Qual Div
Directorate for RD&E

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

AMSAV-EFT
SUBJECT: Revision of Safety of Flight Release for AH-56A Attack Helicopter Evaluation Dated 15 May 1972

Copy furnished
Commanding General
US Army Materiel Command
ATTN: AMCRD-FQ
AMCSF-A

Chief
AAWS Project Manager's Yuma Field Office
ATTN: ASTA Test Team
Yuma, Arizona

CF:
AAWS Project Manager's Ofc

AMSAV-ERA (Mr. J. Marlo)

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

15 MAY 1972

AMSAV-EFT

SUBJECT: Safety of Flight Release for AH-56A Attack Helicopter Evaluation

Commanding Officer
US Army Aviation Systems
Test Activity
ATTN: SAVTE-P

1. Reference is made to:

a. Letter from AMSAV-EF to SAVTE-P, subject: APE 1.3 Safety of Flight Release, dated 1 Sept 71.

b. TWX 09-12, from AMSAV-EFT to SAVTE-P, subject: APE 1.3 Safety of Flight Release, dated 21 Apr 71.

c. TWX 10-03, from AMSAV-EFT to SAVTE-P, subject: AH-56A Safety of Flight Release for APE 1.3 and RDAT I, dated 5 Oct 71.

d. TWX 10-11, from AMSAV-EFT to SAVTE-P, subject: APE 1.3 Safety of Flight Release, dated 8 Oct 71.

e. Letter from AMSAV-EF to SAVTE-P, subject: Revision of APE 1.3 Safety of Flight Release, dated 2 Dec 71.

f. Letter from AMSAV-EFT to SAVTE-P, subject: Safety of Flight Release for AH-56A Attack Helicopter Evaluation, dated 11 Apr 72.

g. Letter from AMSAV-EFT to SAVTE-P, subject: Safety of Flight Release for AH-56A Attack Helicopter Evaluation, dated 18 Apr 72.

2. This letter constitutes a safety of flight release for conduct of AVSCOM/ASTA Project No. 72-08 and supersedes all the references listed above.

3. This flight release is contingent upon the following:

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

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

c. A functioning radio link directly between ground communications and the test aircraft.

d. Proper functioning of flight control augmentation equipment, specifically pitch densitizer and roll compensator units. These units may be turned off for test purposes to determine response characteristics.

4. 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.

(c) Hovering Transitions. Hovering transitions should not be made when power turbine inlet temperatures required for hover exceed 730°C. For hover flight with transition to forward flight, see paragraph 5 for temperature limitations.

(5) Side and Rearward Flight. Sideward and rearward flight limitations are shown in Figure 6, Incl 5.

(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 -5 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 feet 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 feet 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. 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. Touchdown Sink Rates. Touchdown 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.

5. 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 98% (0 to 10 kts and above 50 kts)	yellow band

LIMITATIONS	INSTRUMENT MARKING
Power Turbine RPM con't	
98% to 105% (normal operating range)	green band
105% maximum (power on)	red line
114% overspeed cutoff	(no mark)
Main Rotor RPM	
90% minimum (power off) (95% minimum power on)	red line
90% to 110% normal operating range	green band
110% maximum (power off)	red line
Engine Oil Temperature	
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 7, emergency procedures)	(no mark)
Transmission Oil Temperature	
0 to 113°C normal operating range	green band
113°C maximum continous	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 50 psi, idle	(no mark)

LIMITATIONS	INSTRUMENT MARKING
Engine Oil Pressure con't	
10 psi to 83 psi normal range	green band
50 psi, minimum at 95% gas generator	(no mark)

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

LIMITATIONS	INSTRUMENT MARKING
Transmission Oil Pressure	
70 psi minimum	red line
70 to 110 psi normal operating range	green band
110 psi maximum	red line
Prop. Gearbox Oil Temperature	
121°C maximum	red line
Turbine Inlet Temperature	
300°C to 740°C, normal operating range	green band
740°C, maximum continuous	red line
740°C - 770°C (30 minute limit)	yellow band
770°C - 780°C (10 minute limit)	yellow band
780°C - 785°C (1 minute limit)	yellow band
785°C, inspection limit	red line
720°C, record time above*	blue line

* Engine operating time is limited to a total of 20 hours at PTIT's greater than 740°C. Engine operating time is limited to a total of 40 hours at PTIT's between 720°C and 740°C. Pilot must record operating time at PTIT's greater than 740°C and operating time at PTIT's between 720°C and 740°C.

6. 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.

7. 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 number 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.

8. Notes, 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% 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. Caution. 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. Note. Avoid operation at 40°F or below with visible moisture present.

9. 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.

10. Propeller Blade Angle Limitations:

a. Maximum propeller blade angle is +40.8 degrees.

b. Minimum propeller blade is -12°.

11. 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 operator's manual. The pilot's checklist POMM 55-1520-222-10 CL, (January 1971), Operator's and Crewmember's Checklist, for aircraft serial number 66-8834, with annotated updating furnished by contractor, shall be used.

FOR THE COMMANDER:

Robert D. Hubbard

ROBERT D. HUBBARD
Acting Chief, Flt Std & Qual Div
Directorate for RD&E

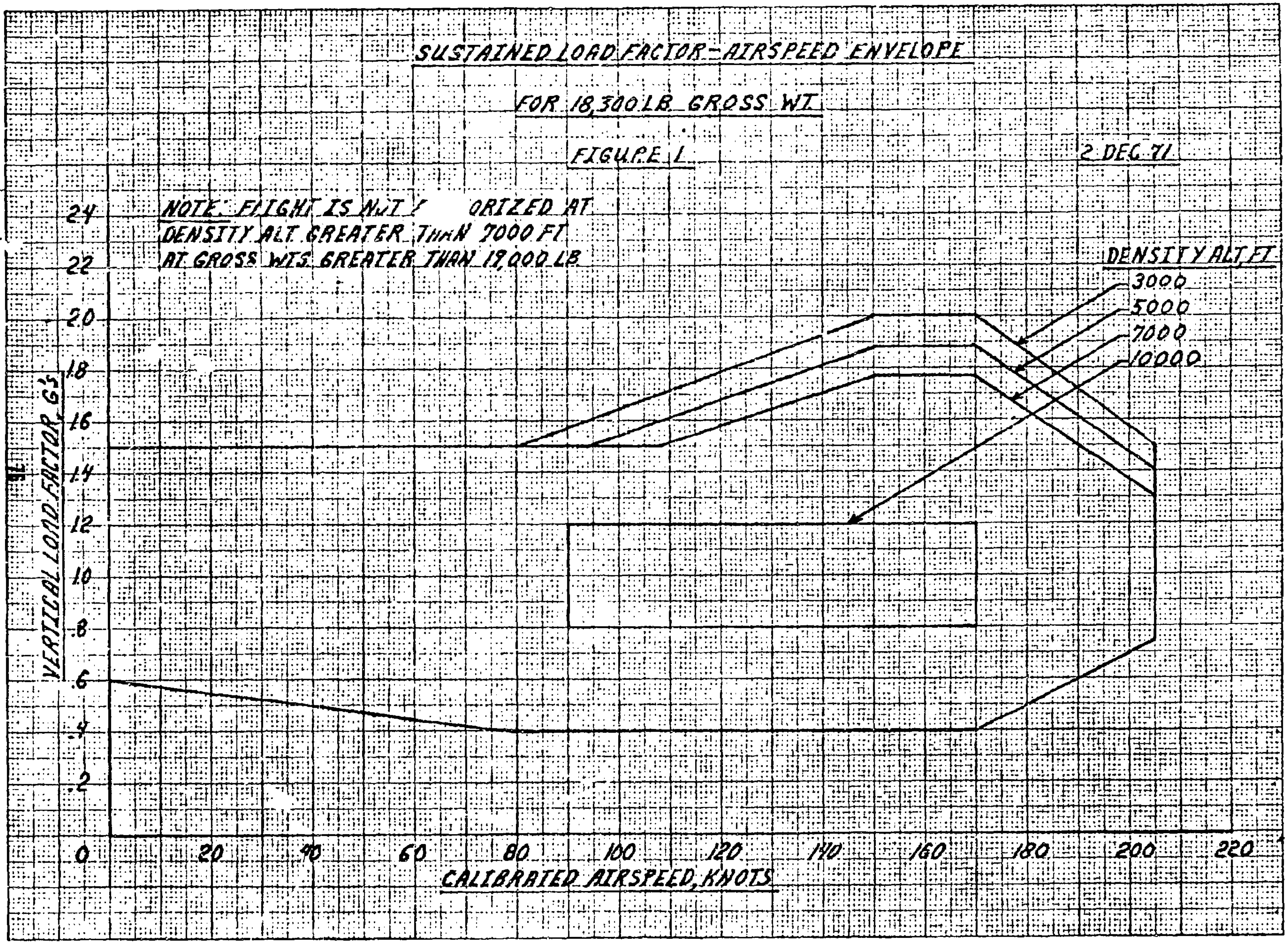
5 Incl
as

AMSAV-EFT [illegible] 1972

SUBJECT: Safety of Flight Release for AH-56A Attack Helicopter Evaluation

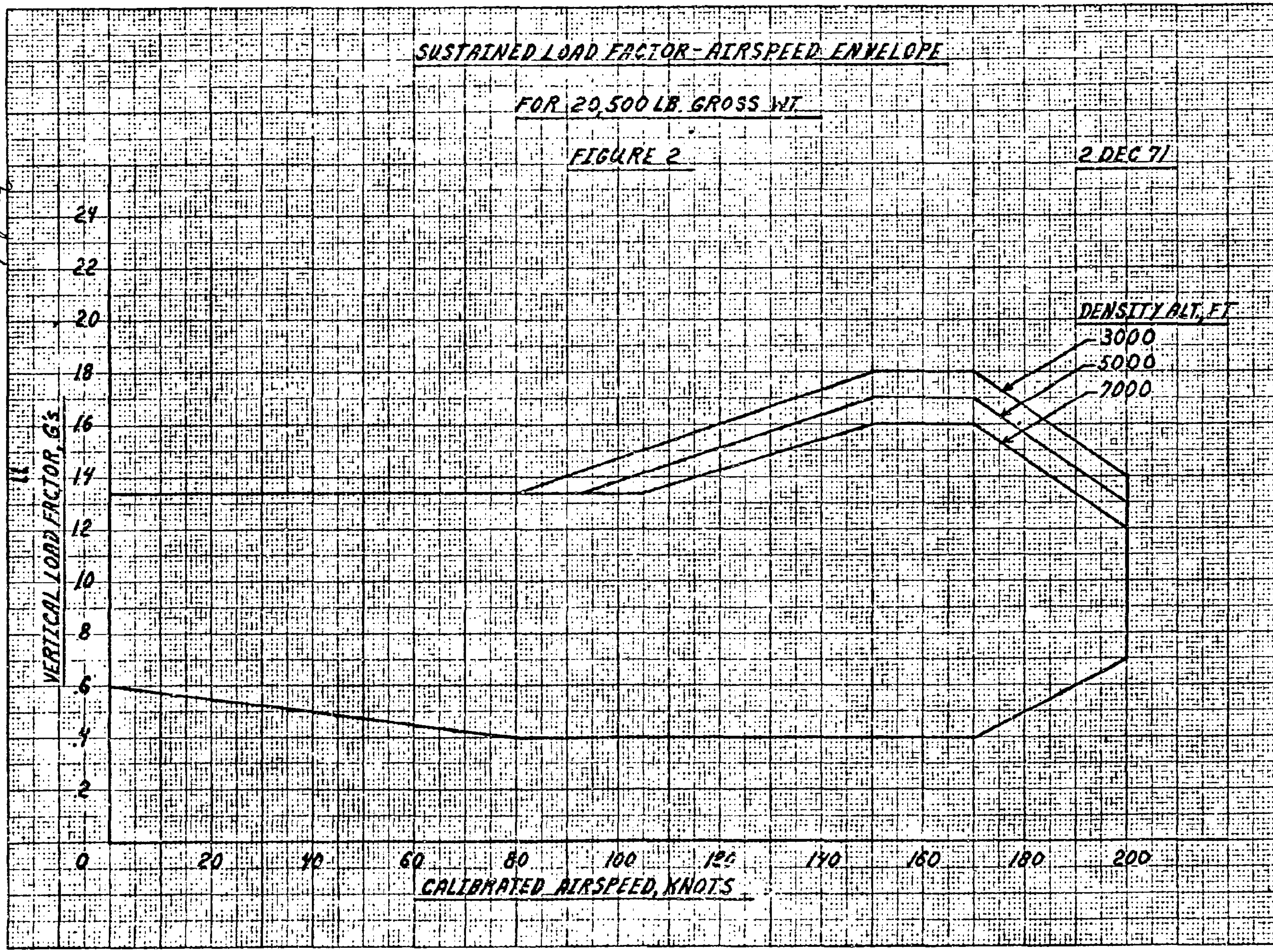
Copy furnished
Commanding General
US Army Materiel Command
ATTN: AMCRD-FQ
AMCSF-A

Chief
AAH - Project Manager's Yuma Field Office
ATTN: ASTA Test Team
Yuma, Arizona


SUSTAINED LOAD FACTOR - AIRSPEED ENVELOPE
FOR 18,300 LB GROSS WT
FIGURE 1
2 DEC 71
NOTE: FLIGHT IS NOT ORIZED AT
DENSITY ALT GREATER THAN 7000 FT
AT GROSS WTS GREATER THAN 19,000 LB
DENSITY ALT, FT
3000
5000
7000
10000
VERTICAL LOAD FACTOR, G'S
24
22
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
.2
0
20
40
60
80
100
120
140
160
180
200
220
CALIBRATED AIRSPEED, KNOTS

76

Incl. 1


SUSTAINED LOAD FACTOR - AIRSPEED ENVELOPE
FOR 20,500 LB. GROSS WT.
FIGURE 2
2 DEC 71
DENSITY ALT, FT
3000
5000
7000
VERTICAL LOAD FACTOR, G's
24
22
20
18
16
14
12
10
8
6
4
2
0
20
40
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED, KNOTS
11

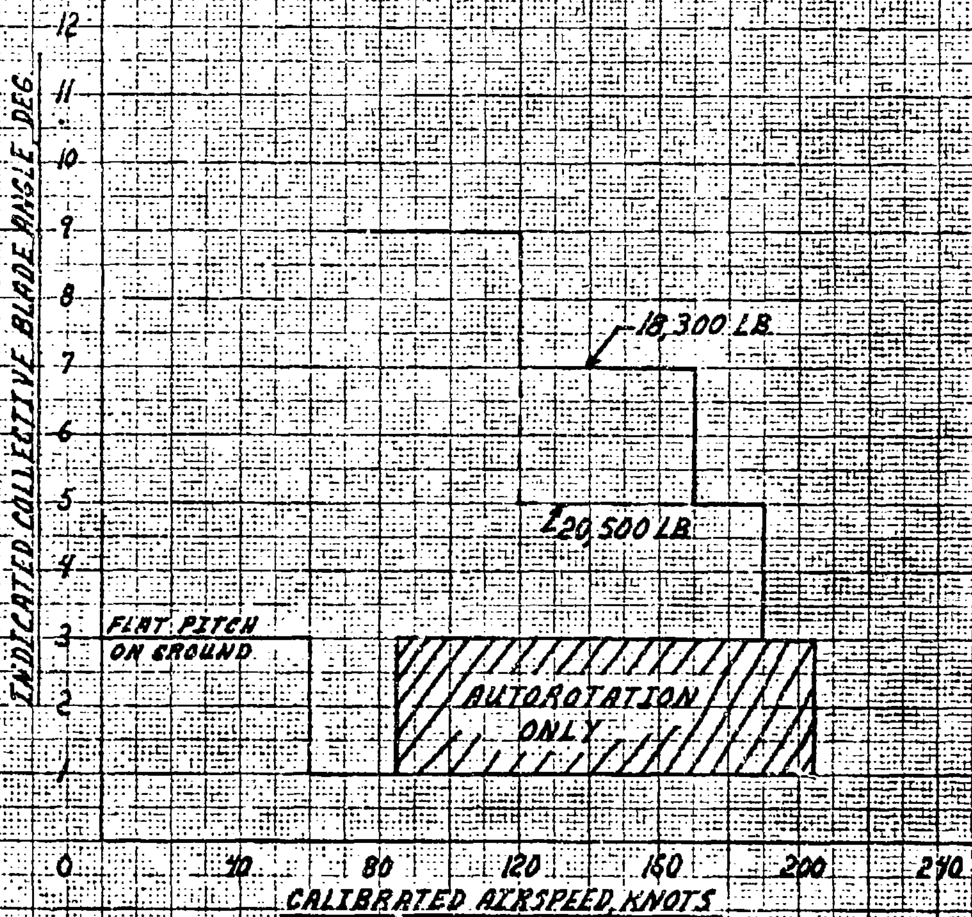
Incl. 1

INDICATED COLLECTIVE BLADE ANGLE - AIRSPEED ENVELOPE

FIGURE 3

2 DEC 71

NOTE: FROM 0 TO 70 KCAS
IND. COLL. BLADE ANGLE
SHALL BE AS REQUIRED



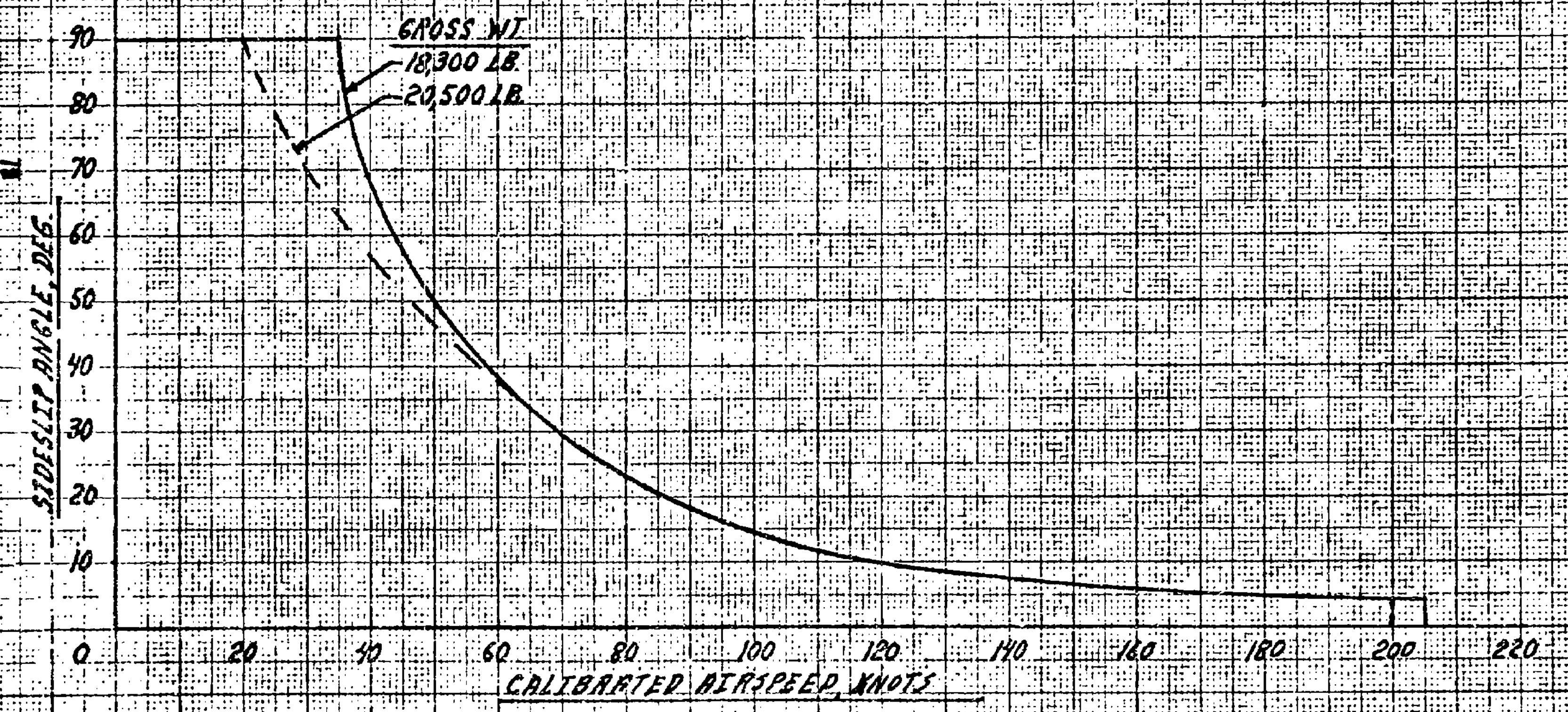
Incl 2

SIDESLIP ANGLE - AIRSPEED ENVELOPE

LEFT OR RIGHT SIDESLIPS

FIGURE 4

2 DEC 71

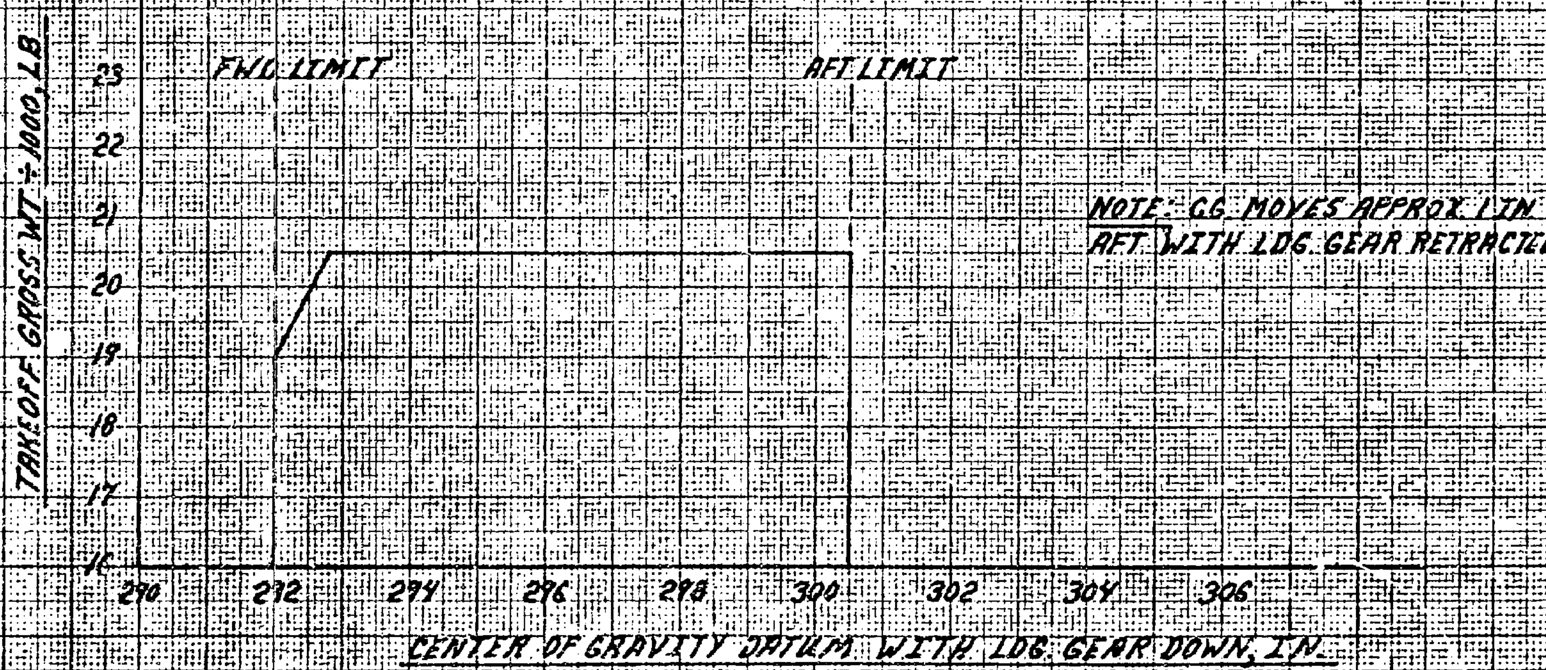


Incl. 3

GROSS WT - CENTER OF GRAVITY ENVELOPE

FIGURE 5

30 AUG 71



Incl. 4

AH-56A SIDEWARD (LEFT & RT) AND REARWARD FLIGHT ENVELOPE

FIGURE 6

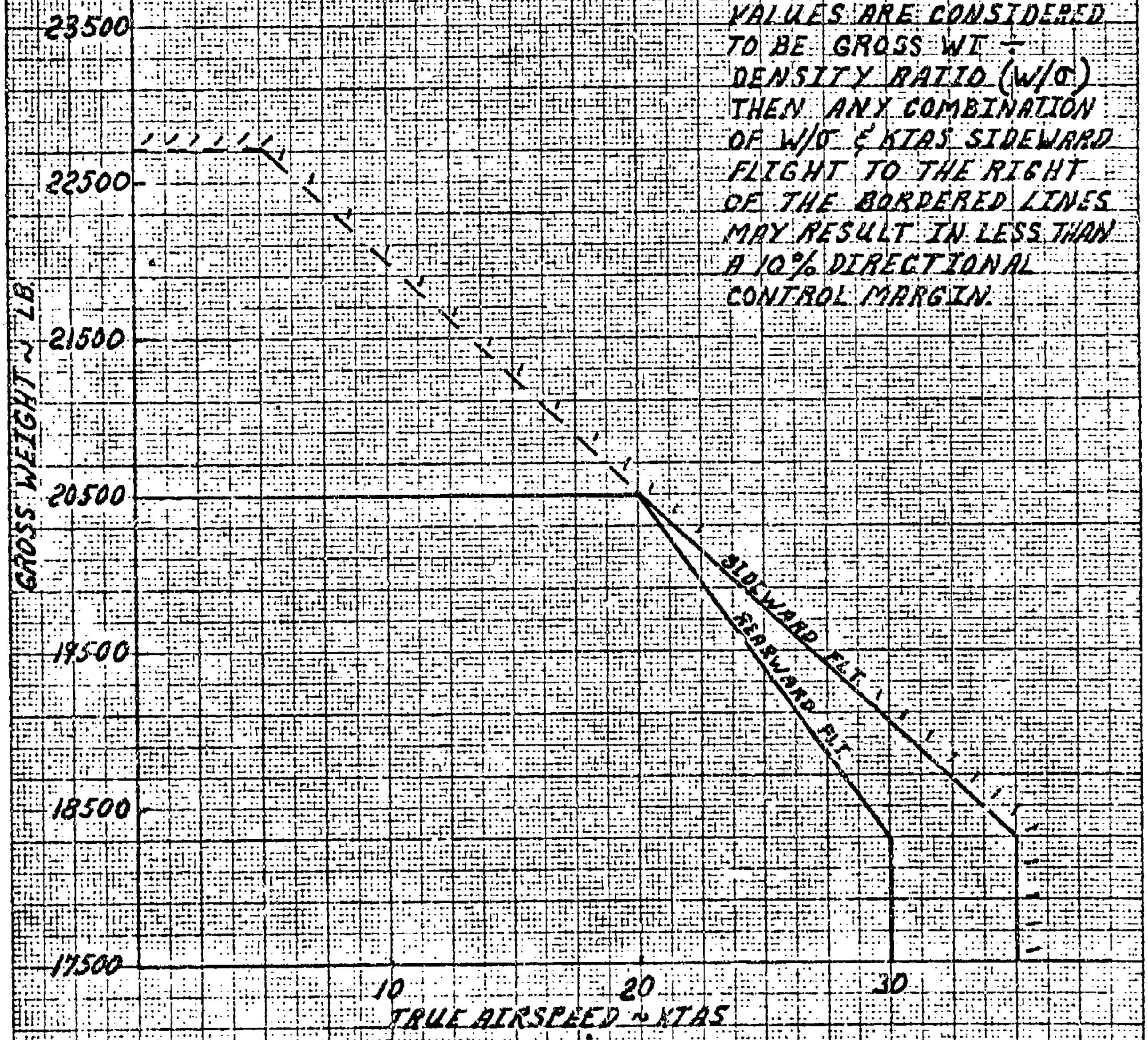
12 APR 72

MAIN ROTOR SPEED = 100%

NOTES:

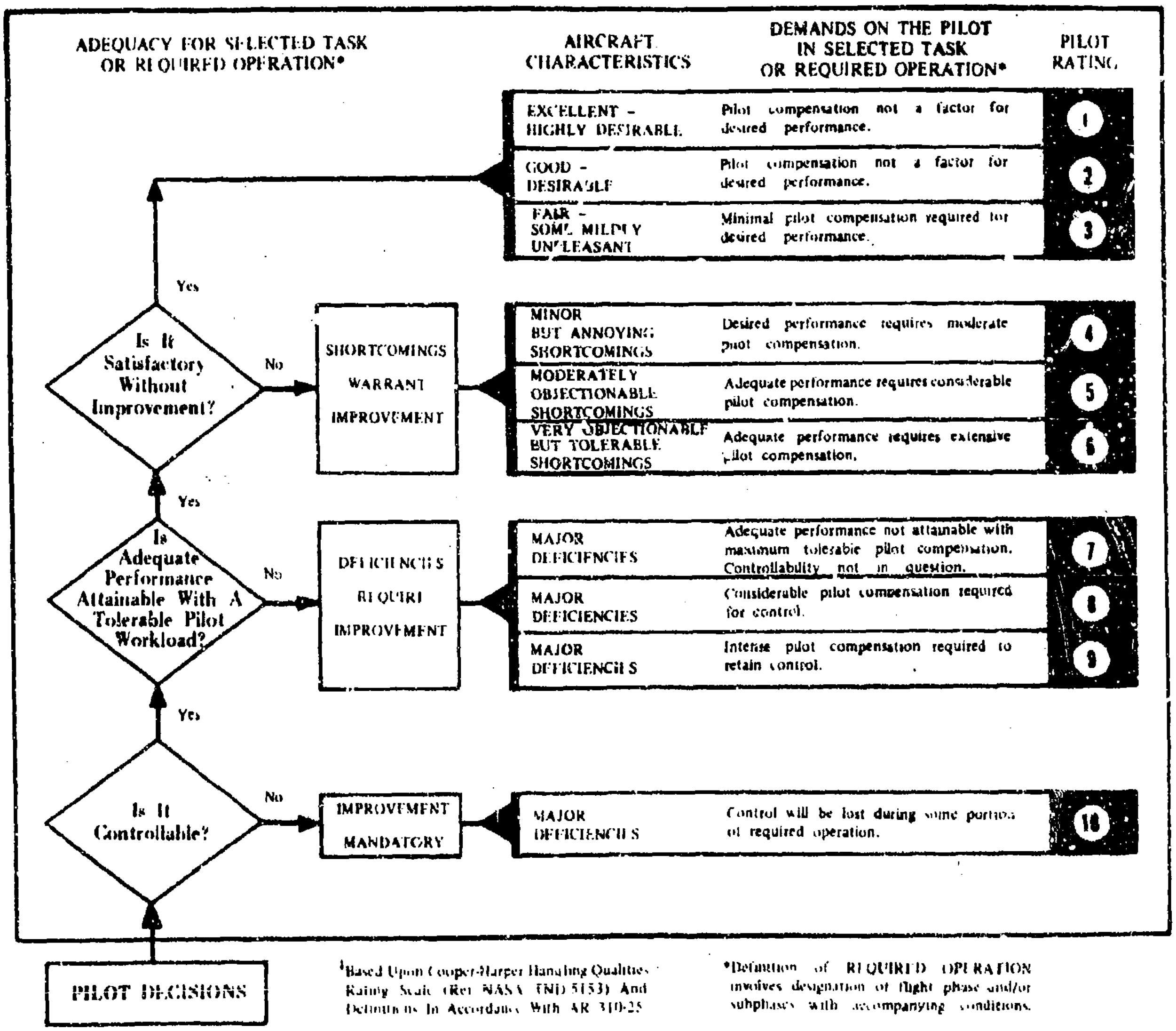
1. THE SOLID LINES SHOW THE MAX. AIRSPEED AUTHORIZED AS A FUNCTION OF GROSS WT.

2. THE SOLID OR DASHED LINE WHEN BORDERED ' ' ' ' ' ' INDICATES THAT IF THE ORDINATE VALUES ARE CONSIDERED TO BE GROSS WT ÷ DENSITY RATIO (W/σ) THEN ANY COMBINATION OF W/σ & KTAS SIDEWARD FLIGHT TO THE RIGHT OF THE BORDERED LINES MAY RESULT IN LESS THAN A 10% DIRECTIONAL CONTROL MARGIN.



Incl 5

APPENDIX F. 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 G. DATA ANALYSIS METHODS

Introduction

1. This appendix contains some of the test techniques and data reduction and analysis methods used to evaluate the AH-56A. The topics discussed include:

   a. Shaft horsepower required.

   b. Hover performance.

   c. Tail rotor performance.

   d. Level flight performance and specific range.

   e. Forward flight acceleration and deceleration performance.

General

2. The nondimensional equations used for hover and level flight performance analysis are defined as follows:

   a. Coefficient of power (C_P):

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

   b. Coefficient of thrust (C_T):

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

   c. Advance ratio (μ):

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

where: SHP = Engine output shaft horsepower

550 = Conversion factor (ft-lb/sec per SHP)

ρ = Air density (slug/ft^3)

A = Main rotor disc area (ft^2)

Ω = Main rotor angular velocity (radians)

R = Main rotor radius (ft)

W = Aircraft gross weight (lb)

V_T = True airspeed (ft/sec)

Shaft Horsepower Required

3. The engine output shaft torque was determined by measuring the torsional strain of the engine output shaft. From laboratory static calibrations, the shaft's torsional strain was related to the applied torque. Dynamic, zero-torque reference points were obtained during autorotational descents to correlate with static calibrations. The shaft strain measuring system was electrical and its output was displayed on an indicator calibrated in increments of shaft horsepower at 100 percent power turbine speed (N_f). (The system also had outputs on indicators calibrated in percent of torque but the gage increments were too coarse for test purposes). Corrections to other than 100 percent N_f were made by simply multiplying the indicated shaft horsepower (corrected for instrument error) times the ratio of actual N_f to 100 percent ($\frac{N_f}{100}$). An alternate method of obtaining shp (which was used as a check of the above system) was to measure main rotor and tail drive torques using strain gages, and calculate engine horsepower required (assuming a constant 90-shp transmission loss).

4. Tail rotor shaft horsepower was determined from the following equation:

$$SHP = \frac{2\pi x GR x N_f x Q_{TR}}{33{,}000} \qquad (4)$$

where: GR = Ratio of tail rotor rotational speed to N_f

N_f = Power turbine rotational speed (RPM)

Q_{TR} = Tail rotor shaft torque (ft-lb)

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

Tail rotor torque was measured using strain gages.

Hover Performance

5. Hover performance was determined in-ground-effect (IGE) by stabilizing the aircraft at a 10-foot wheel height and recording performance data. The aircraft gross weight and main rotor speed (N_R) were varied from point to point and the tests were conducted at two density altitudes in order to get maximum possible variiation of C_T (equation (2)). The objective of the test is to determine the variation of C_p (equation (1)) with C_T in order to define the aircraft

hover capability. Out-of-ground-effect hover performance was determined in the same manner except that the wheel height was at least 80 feet for each point. Height reference for the tests was a measured, weighted cable suspended from the aircraft.

Tail Rotor Performance

6. During the hover performance tests, tail rotor performance parameters were also recorded. Terms in equations (1) and (2 which apply to the main rotor were replaced by tail rotor parameters to nondimensionalize tail rotor performance. The terms redefined are as follows:

SHP = Tail rotor shaft horsepower (equation (4))

A = Tail rotor disc area (ft^2)

Ω = Tail rotor angular velocity (radians/sec)

R = Tail rotor radius (ft)

W = Tail rotor thrust (lb)

Tail rotor thrust was determined from the following equation:

$$W = \frac{Q_{MR}}{x_t} \tag{5}$$

where: Q = Main rotor shaft torque (ft-lb)

x_t = Perpendicular distance between center lines of main and tail rotor shafts (ft)

Level Flight Performance and Specific Range

7. Level flight performance was determined by stabilizing the aircraft at zero sideslip at increments of airspeed from maximum level flight speed (V_H) to the minimum airspeed at which level flight could be maintained without increasing collective blade angle. Constant altitude was maintained during each stabilized point. External configuration, C_T, and collective blade angle were held constant for each sweep of airspeed (speed power polar). C_T was held constant by keeping N_R constant and increasing altitude between data points to allow for fuel burn-off (i.e. to maintain a constant ratio of aircraft weight (W) to air density ratio (σ)). At each stabilized point, performance parameters were recorded. Comparisons were made using equations (1), (2), and (3).

8. Test-day level-flight power was corrected to standard-day conditions by assuming that the test-day dimensionless parameters, C_{P_t}, C_{T_t}, and μ_t, are independent of atmospheric conditions. Consequently, the standard day dimensionless parameters, C_{P_S}, C_{T_S}, and μ_S, are identical to C_{P_t}, C_{T_t}, and μ_t, respectively. A corollary to the above assumption relates:

$$\rho_s = \rho_t \left(\frac{W_s}{W_t}\right) \left(\frac{N_{R_t}}{N_{R_S}}\right)^2 \qquad (6)$$

where: ρ = Air density (slug/ft^3)

W = Aircraft gross weight (lb)

Subscript t = Test day

Subscript s = Standard day

9. Equation (6) defines the standard-day density (ρ_s) which is required for presentation of test-day data at a standard gross weight (W_s) and the aim main rotor speed N_{R_S}. The standard gross weight is determined by averaging the gross weights at individual test points for an entire speed-power.

10. From the definition of C_P (equation (1)) the following relationship can be derived:

$$SHP_s = SHP_t \times \frac{\rho_s}{\rho_t} \qquad (7)$$

This relationship defines the power required to fly level at the same thrust and power coefficients and advance ratios as on the test day but under standard-day conditions. Each test point was corrected in this fashion to standard-day conditions at the target gross weight and N_R.

11. Specific range was calculated using the level flight performance curves and the specification installed-engine fuel-flow characteristics as follows:

$$NAMPP = \frac{V_T}{W_f} \qquad (8)$$

where: NAMPP = Nautical air miles per pound of fuel (naut mi/lb)

V_T = True airspeed (KT)

W_f = Fuel flow (lb/hr)

APPENDIX H. TEST INSTRUMENTATION

1. All test instrumentation was installed and maintained by the contractor during this evaluation. Data were recorded on two oscillographs and a photographic automatic observer panel. Some data were hand recorded from the two cockpit instrument panels. Additionally, 18 parameters could be monitored almost in real time via a telemetry link. Included in the instrumentation package were the following:

PILOT PANEL

Airspeed (boom)
Altitude (boom)
Rotor speed
Engine torque
Turbine inlet temperature
Longitudinal stick position
Lateral stick position
Pedal position
Collective blade angle
Center-of-gravity normal acceleration
Angle of sideslip
Angle of attack
Propeller blade angle
Total air temperature
Gas producer speed
Power turbine speed
Fuel quantity
Vertical speed
Directional gyro
Pilot event
Time of day
Correlation counter

ENGINEER's (COPILOT) PANEL

Airspeed (boom)
Altitude (boom)
Rotor speed
Engine torque
Turbine inlet temperature
Collective blade angle
Center-of-gravity normal acceleration
Propeller blade angle
Fuel used
Total air temperature

Engineer event
Time of day
Correlation counter

PHOTOPANEL

Airspeed (boom and ship's system)
Altitude (boom and ship's system)
Free air temperature
Rotor speed
Gas generator speed
Power turbine speed
Fuel used
Fuel flow
Fuel temperature
Engine torque
Turbine inlet temperature
Time of day
Pilot and engineer event lights
Vertical speed
Correlation counter

OSCILLOGRAPH #1

Control positions:
  Longitudinal cyclic
  Lateral cyclic
  Collective
Control force:
  Longitudinal cyclic
  Lateral cyclic
Aircraft attitude:
  Pitch
  Roll
Aircraft angular rate:
  Pitch
  Roll
Aircraft angular acceleration:
  Pitch
  Roll
Center-of-gravity normal acceleration (filtered at 2 Hz)
Angle of attack
Angle of sideslip
Main rotor index
Pilot event
Engineer event
Main 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 shaft bending at zero degrees
Main rotor shaft bending at 90 degrees
Main rotor pitch link axial load
Main rotor gyro drive torque
Swashplate collective position
Swashplate roll position
Swashplate pitch position
Pitch load below swashplate
Collective load below swashplate
Correlation counter

OSCILLOGRAPH #2

Control positions:
- Pedal
- Pusher propeller blade angle

Control force:
- Pedal

Aircraft angular rate:
- Yaw

Aircraft angular acceleration:
- Yaw

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 blade angle
Tail rotor shaft torque
Tail rotor index
Propeller index
Main rotor index
Correlation counter
Pilot event

TELEMETRY

2. 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 and oscilloscope in real time, as well as being recorded on oscillograph and magnetic tape.

APPENDIX I. TEST DATA

This appendix contains test data obtained during the Attack Helicopter Evaluation of the AH-56A Cheyenne Compound Helicopter.

INDEX

FIGURE 7
SUMMARY HOVER CAPABILITY
AH-56A USA S/N 66-8834
WHEEL HEIGHT - 10 FEET

MAXIMUM RATED POWER

NOTES:
1. SPECIFICATION POWER BASED ON GENERAL ELECTRIC COMPUTER DECK NO. P643051
2. HOVER DATA DERIVED FROM FIGURES 3 AND 4
3. ROTOR SPEED = 246 RPM
4. WIND LESS THAN 3 KNOTS
5. FREE FLIGHT HOVER
6. VERTICAL DISTANCE FROM BOTTOM OF WHEEL TO CENTER OF FIXED HUB = 14 FEET
7. BOTH CURVES EXTRAPOLATED 5 TO 15% BEYOND RANGE OF TEST DATA

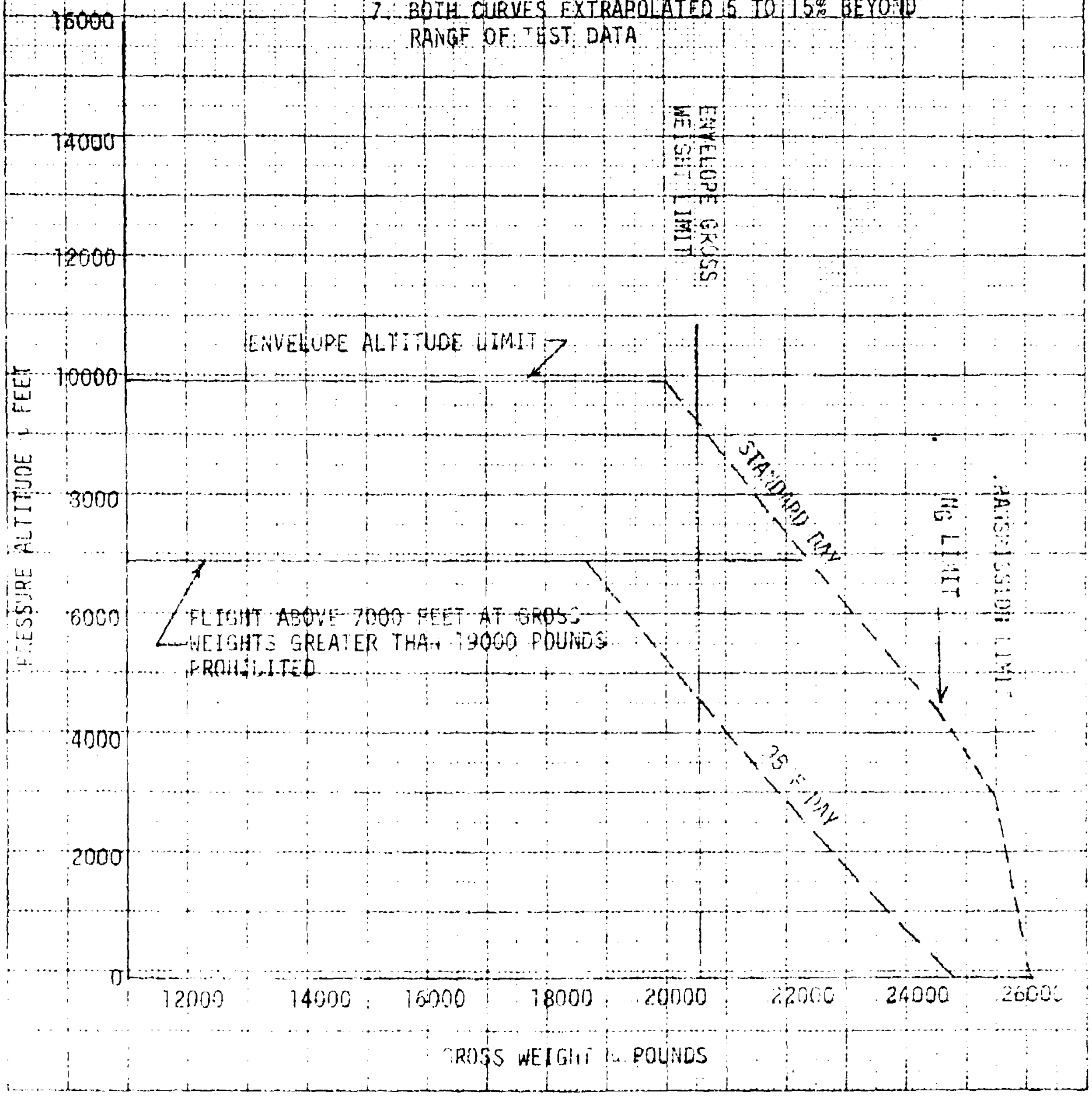
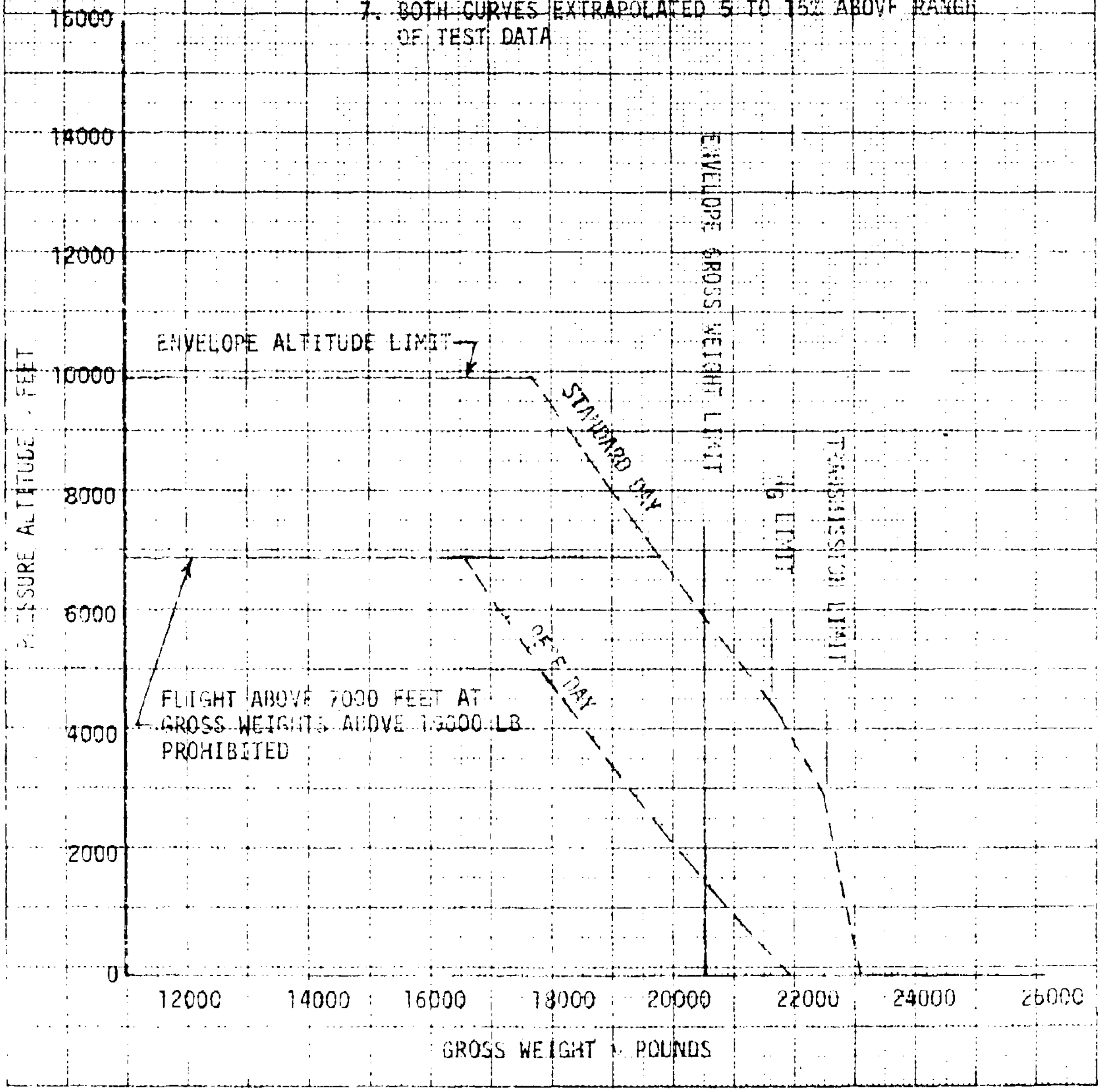


FIGURE 2
SUMMARY HOVER CAPABILITY
AH-56A USA S/N 66-8834
OUT OF GROUND EFFECT

MAXIMUM RATED POWER

NOTES 1. SPECIFICATION POWER BASED ON GENERAL ELECTRIC COMPUTER DECK NO. P643051
2. HOVER DATA DERIVED FROM FIGURES 4 AND 40
3. ROTOR SPEED = 246 RPM
4. WIND LESS THAN 3 KNOTS
5. FREE FLIGHT HOVER
6. VERTICAL DISTANCE FROM BOTTOM OF WHEEL TO CENTER OF FIXED HUB = 14 FEET
7. BOTH CURVES EXTRAPOLATED 5 TO 15% ABOVE RANGE OF TEST DATA



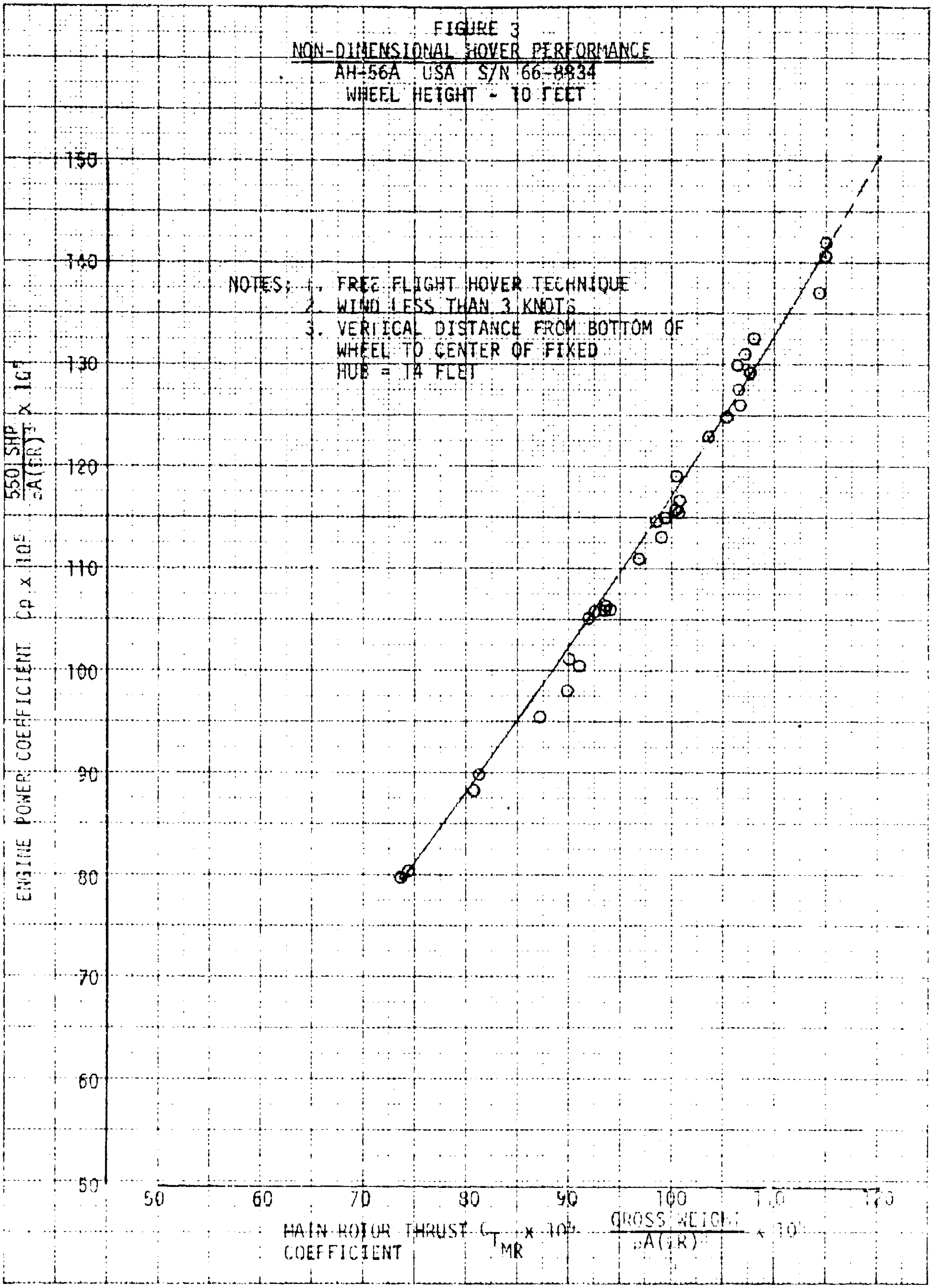


FIGURE 3
NON-DIMENSIONAL HOVER PERFORMANCE
AH-56A USA S/N 66-8834
WHEEL HEIGHT - 10 FEET
NOTES: 1. FREE FLIGHT HOVER TECHNIQUE
2. WIND LESS THAN 3 KNOTS
3. VERTICAL DISTANCE FROM BOTTOM OF WHEEL TO CENTER OF FIXED HUB = 14 FEET
ENGINE POWER COEFFICIENT Cp x 10^5 550 SHP / ρA(ΩR)^3 x 10^5
150
140
130
120
110
100
90
80
70
60
50
50
60
70
80
90
100
110
120
MAIN ROTOR THRUST COEFFICIENT C_TMR x 10^5 GROSS WEIGHT / ρA(ΩR)^2 x 10^5

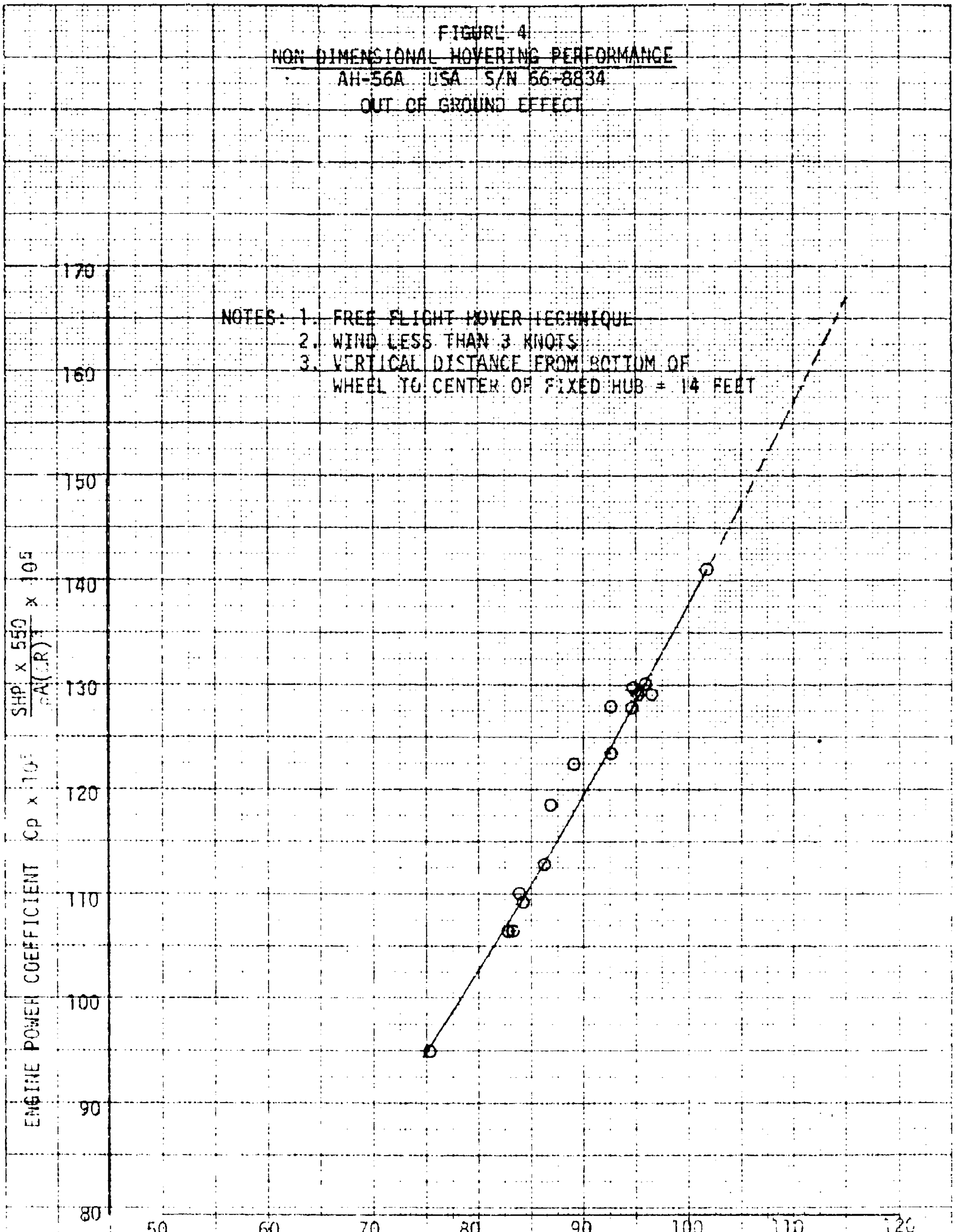
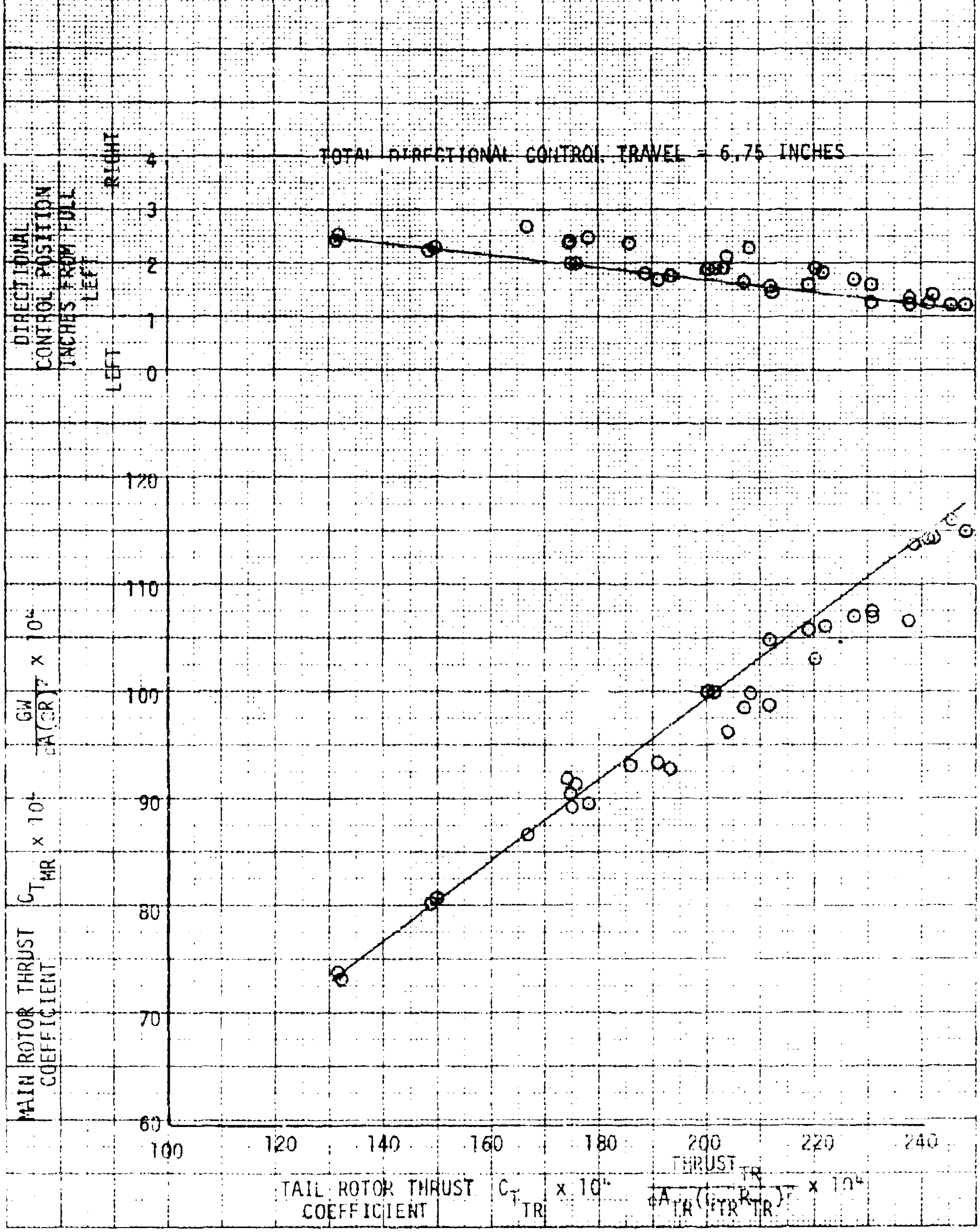


FIGURE 4
NON DIMENSIONAL HOVERING PERFORMANCE
AH-56A USA S/N 66-8834
OUT OF GROUND EFFECT
NOTES: 1. FREE FLIGHT HOVER TECHNIQUE
2. WIND LESS THAN 3 KNOTS
3. VERTICAL DISTANCE FROM BOTTOM OF WHEEL TO CENTER OF FIXED HUB = 14 FEET
ENGINE POWER COEFFICIENT Cp x 10^5 = SHP x 550 / ρA(ΩR)^3 x 10^5
MAIN ROTOR THRUST COEFFICIENT CT_MR x 10^5 = GW / ρA(ΩR)^2 x 10^5
170
160
150
140
130
120
110
100
90
80
50
60
70
80
90
100
110
120

FIGURE 5
NON-DIMENSIONAL TAIL ROTOR PERFORMANCE
AH-56A USA S/N 66-8834
WHEEL HEIGHT - 10 FEET



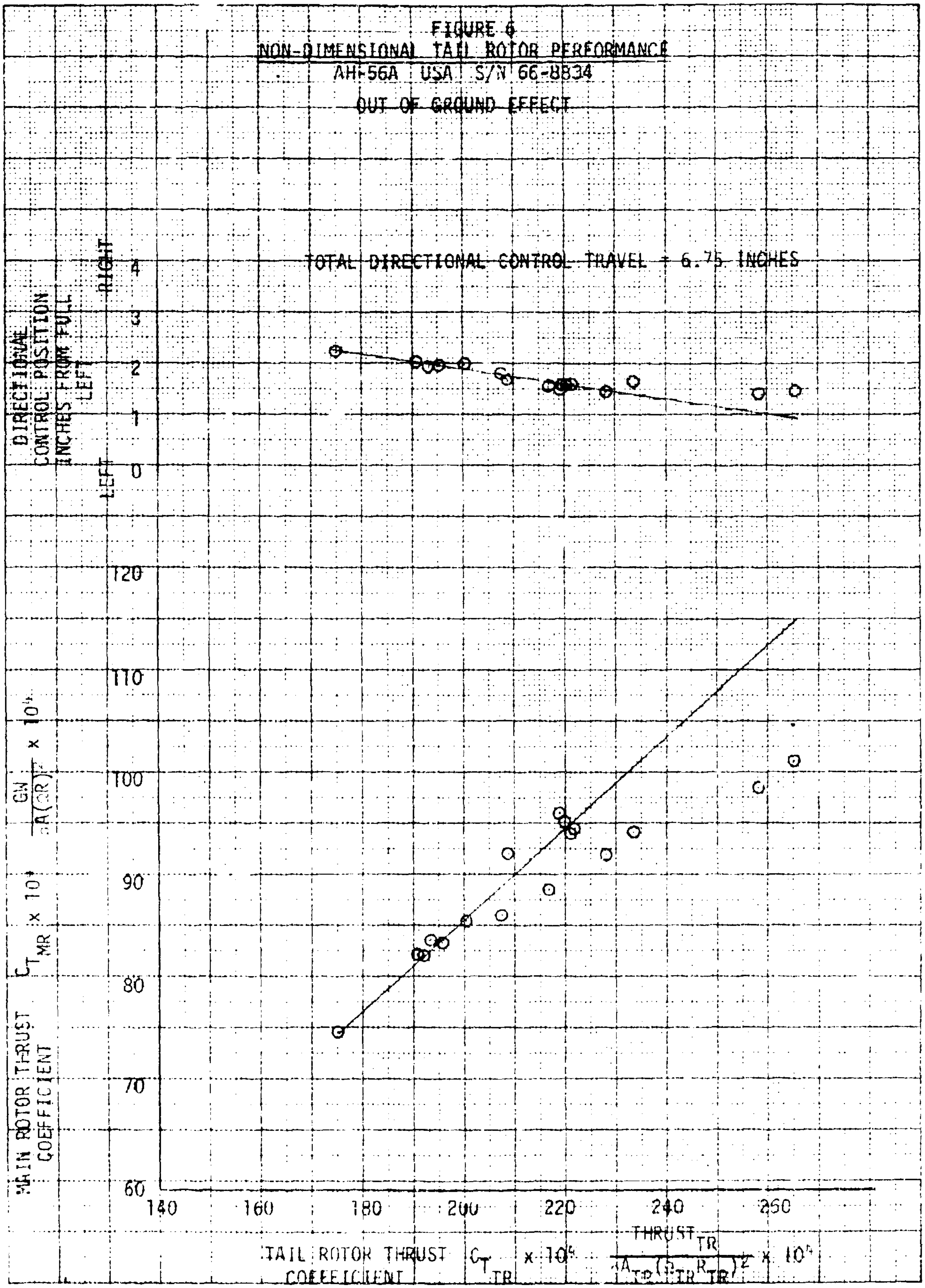
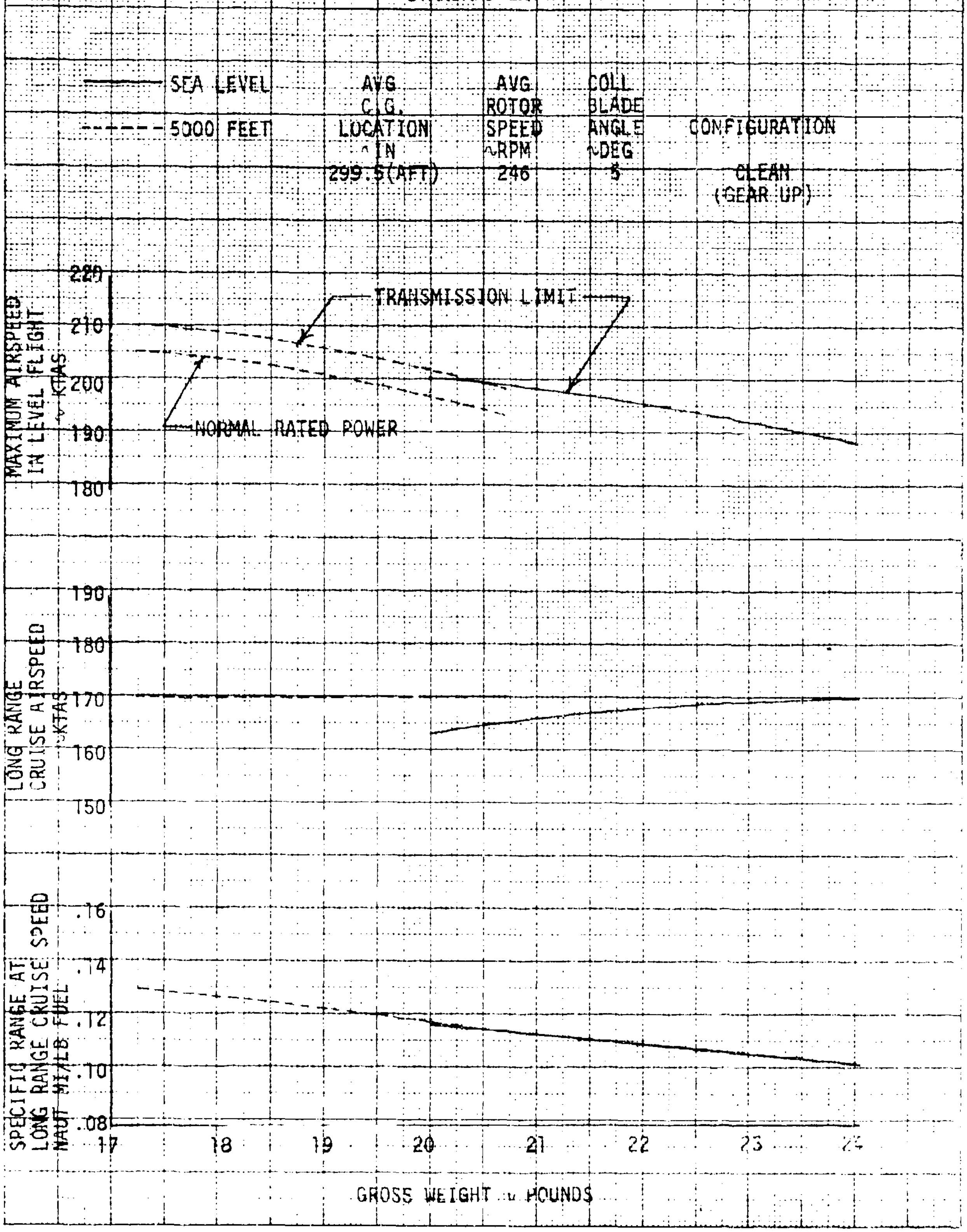


FIGURE 6
NON-DIMENSIONAL TAIL ROTOR PERFORMANCE
AH-56A USA S/N 66-8834
OUT OF GROUND EFFECT
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75 INCHES
DIRECTIONAL CONTROL POSITION INCHES FROM FULL LEFT
RIGHT
LEFT
0
1
2
3
4
MAIN ROTOR THRUST COEFFICIENT
60
70
80
90
100
110
120
140
160
180
200
220
240
260
TAIL ROTOR THRUST COEFFICIENT

FIGURE 7
LONG RANGE SUMMARY
AH-56A USA S/N 66-8834
STANDARD DAY



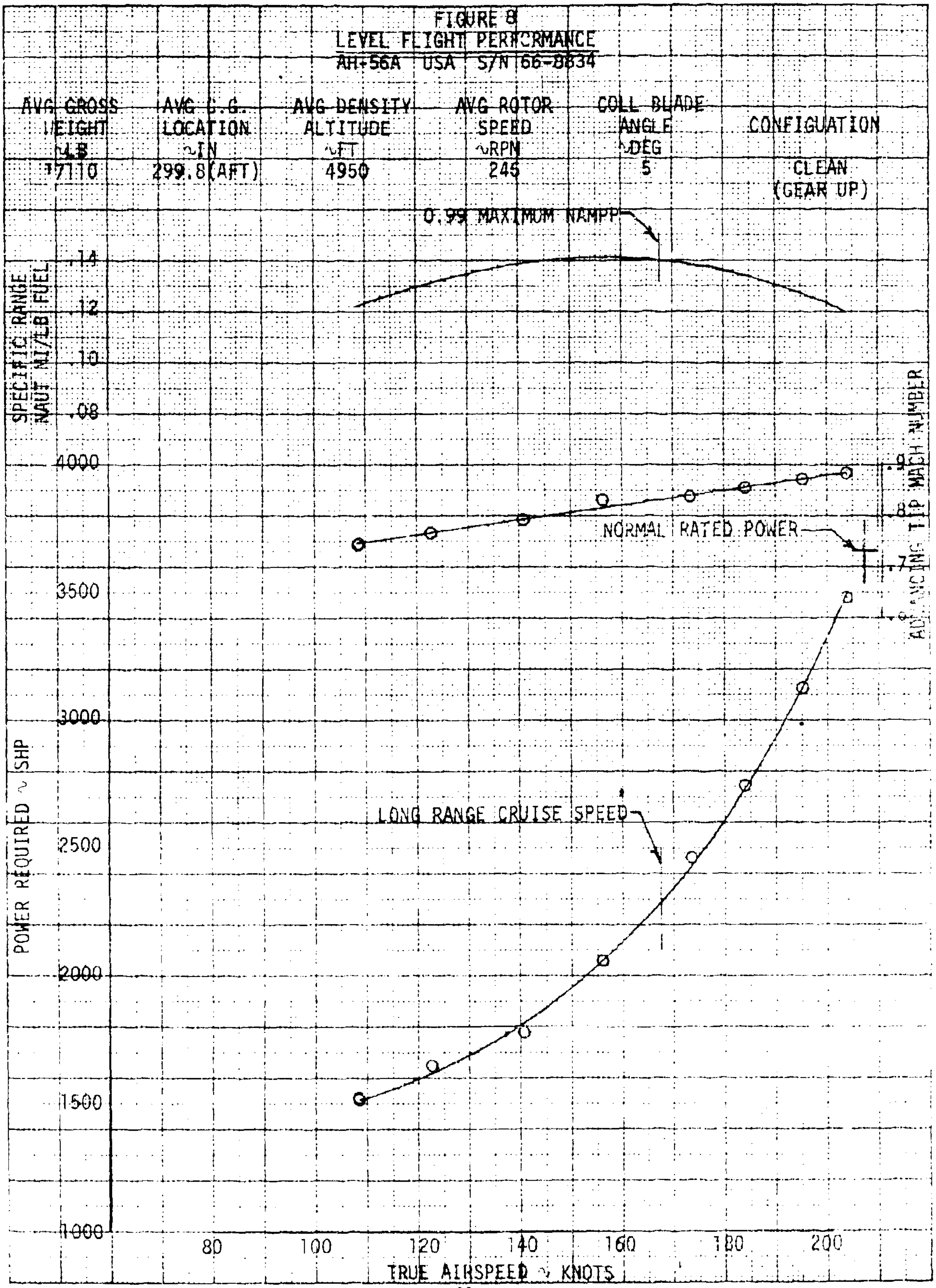


FIGURE 8
LEVEL FLIGHT PERFORMANCE
AH-56A USA S/N 66-8834

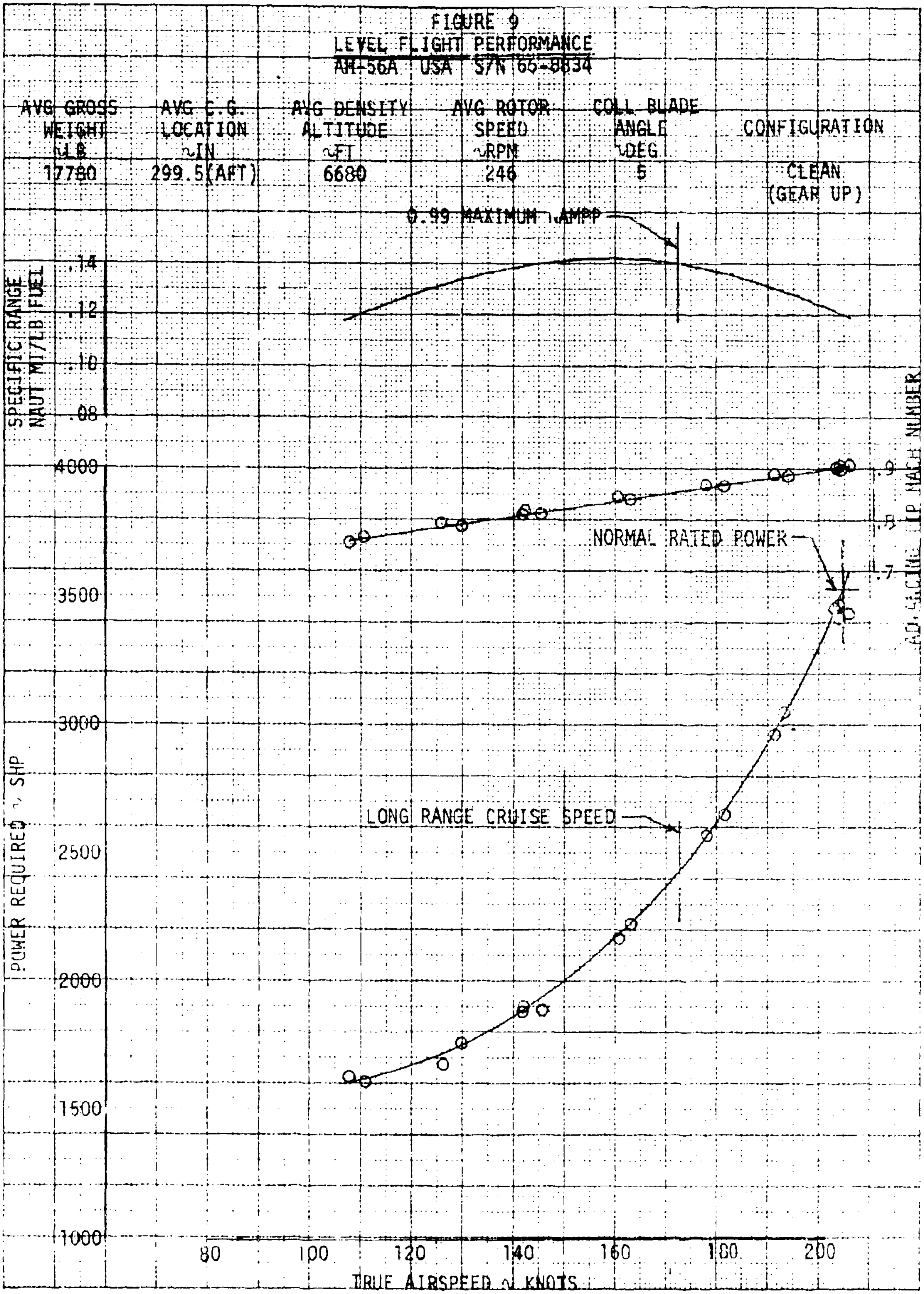
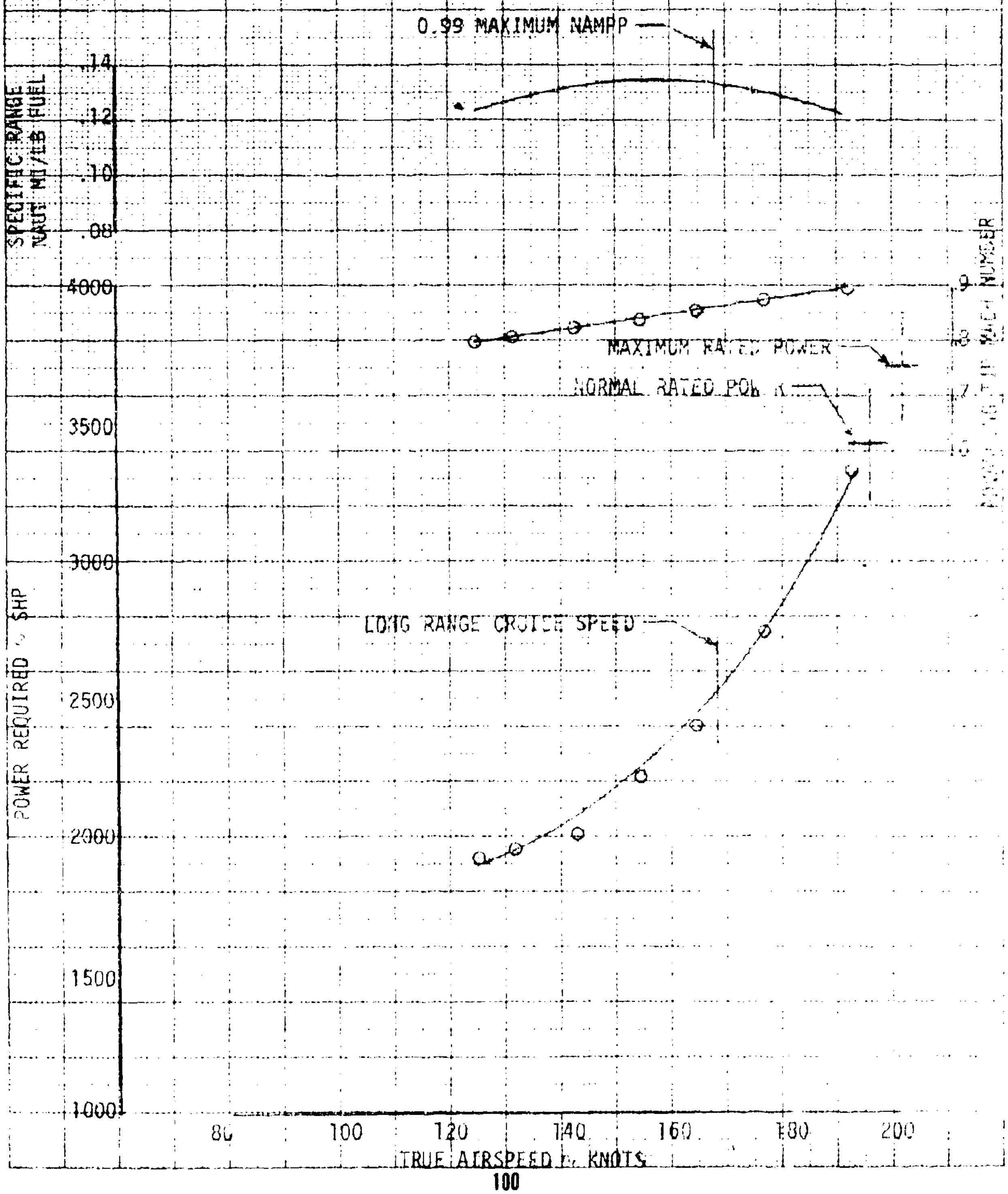


FIGURE 9
LEVEL FLIGHT PERFORMANCE
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB
17780
AVG C.G. LOCATION ~IN
299.5(AFT)
AVG DENSITY ALTITUDE ~FT
6680
AVG ROTOR SPEED ~RPM
246
COLL BLADE ANGLE ~DEG
5
CONFIGURATION
CLEAN (GEAR UP)
0.99 MAXIMUM
SPECIFIC RANGE NAUT MI/LB FUEL
.14
.12
.10
.08
ADVANCING TIP MACH NUMBER
.9
.8
.7
NORMAL RATED POWER
LONG RANGE CRUISE SPEED
POWER REQUIRED ~ SHP
4000
3500
3000
2500
2000
1500
1000
80
100
120
140
160
180
200
TRUE AIRSPEED ~ KNOTS

FIGURE 10
LEVEL FLIGHT PERFORMANCE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG ROTOR SPEED ~RPM	COLL BLADE ANGLE DEG	CONFIGURATION
19000	298.3(AFT)	7730	246	5	CLEAN (GEAR UP)



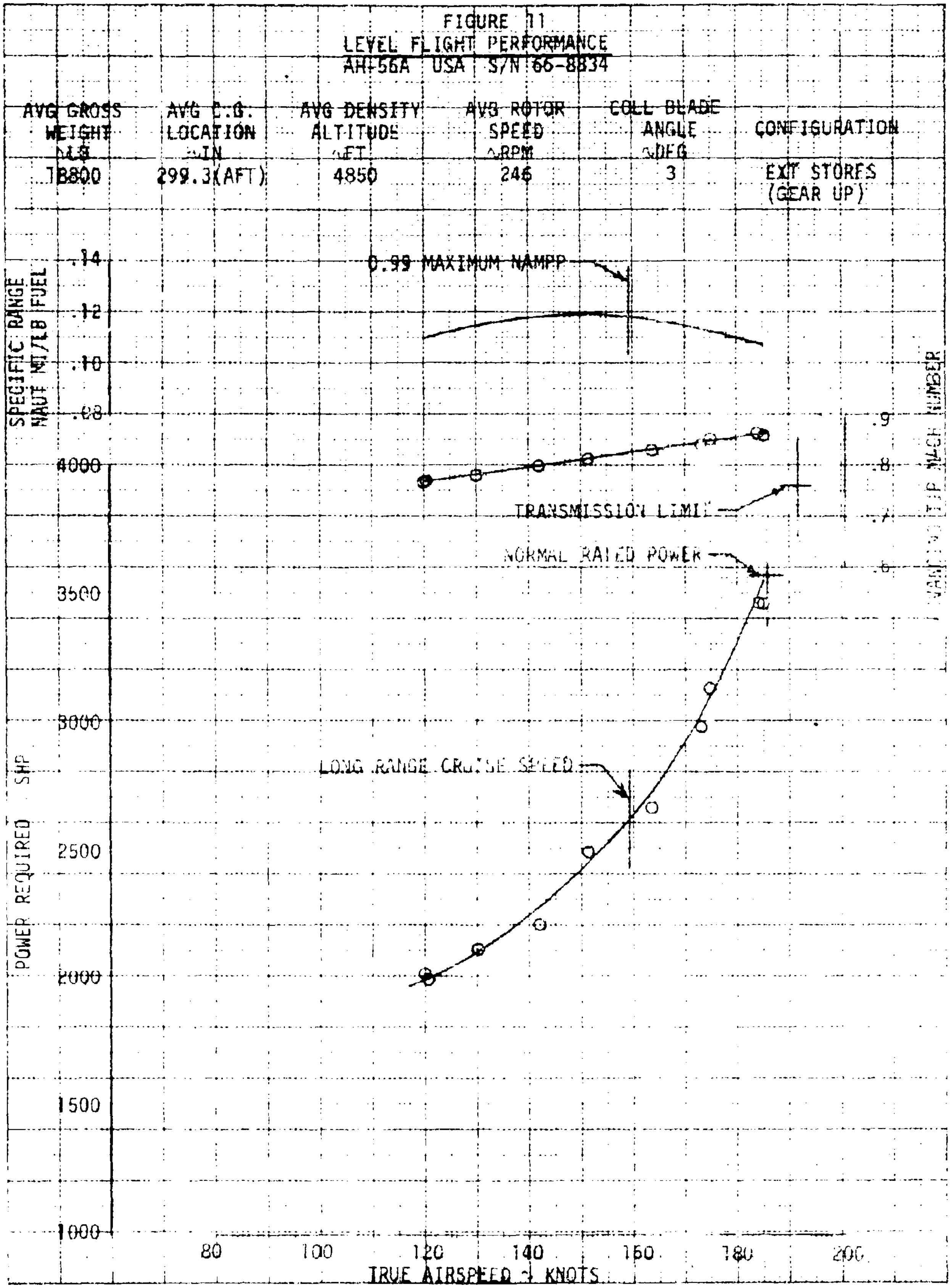
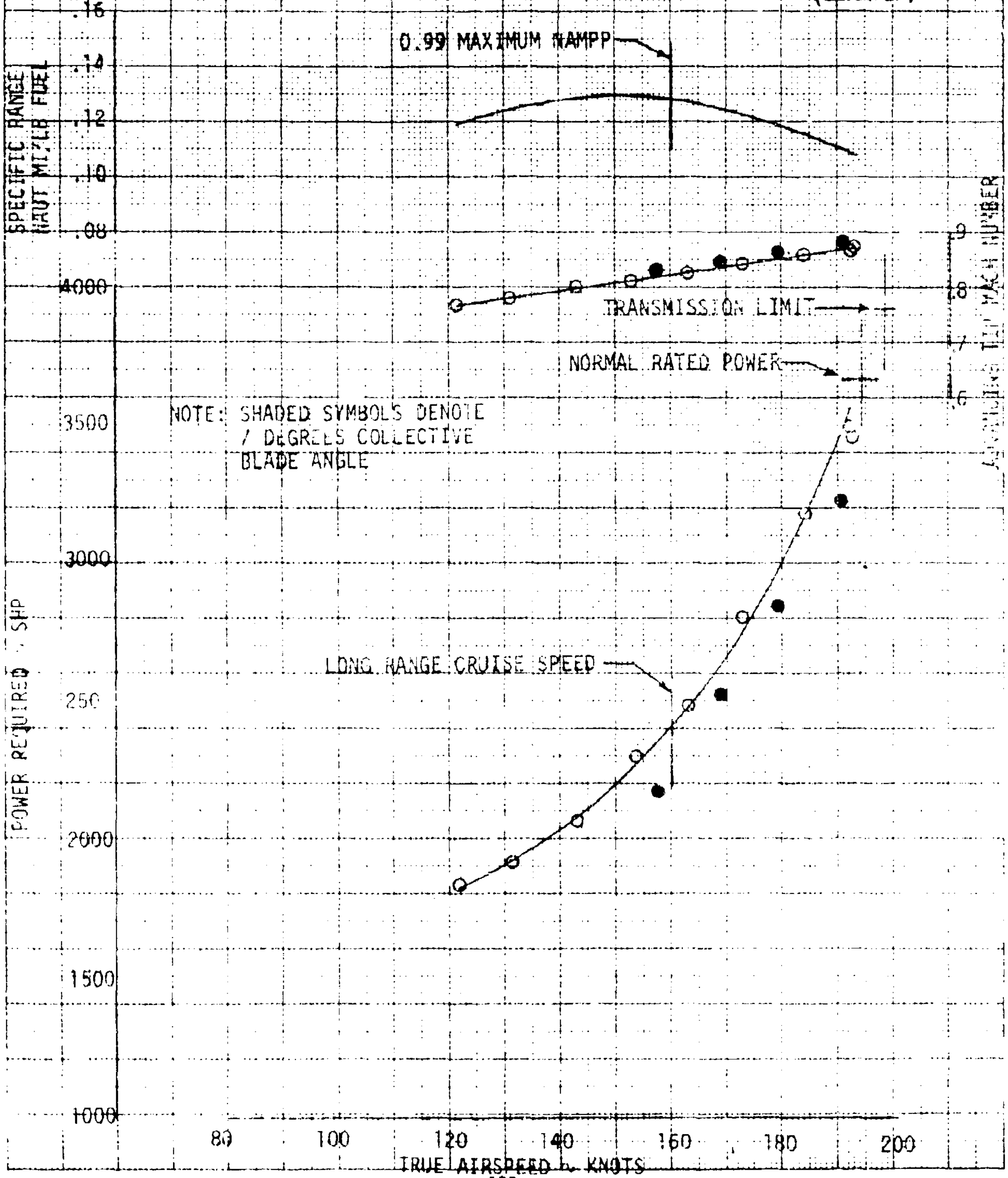


FIGURE 11
LEVEL FLIGHT PERFORMANCE
AH-56A USA S/N 66-8834

FIGURE 12
LEVEL FLIGHT PERFORMANCE
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CONFIGURATION
18800	299.2(AFT)	4850	246	5	EXT STORES (GEAR UP)



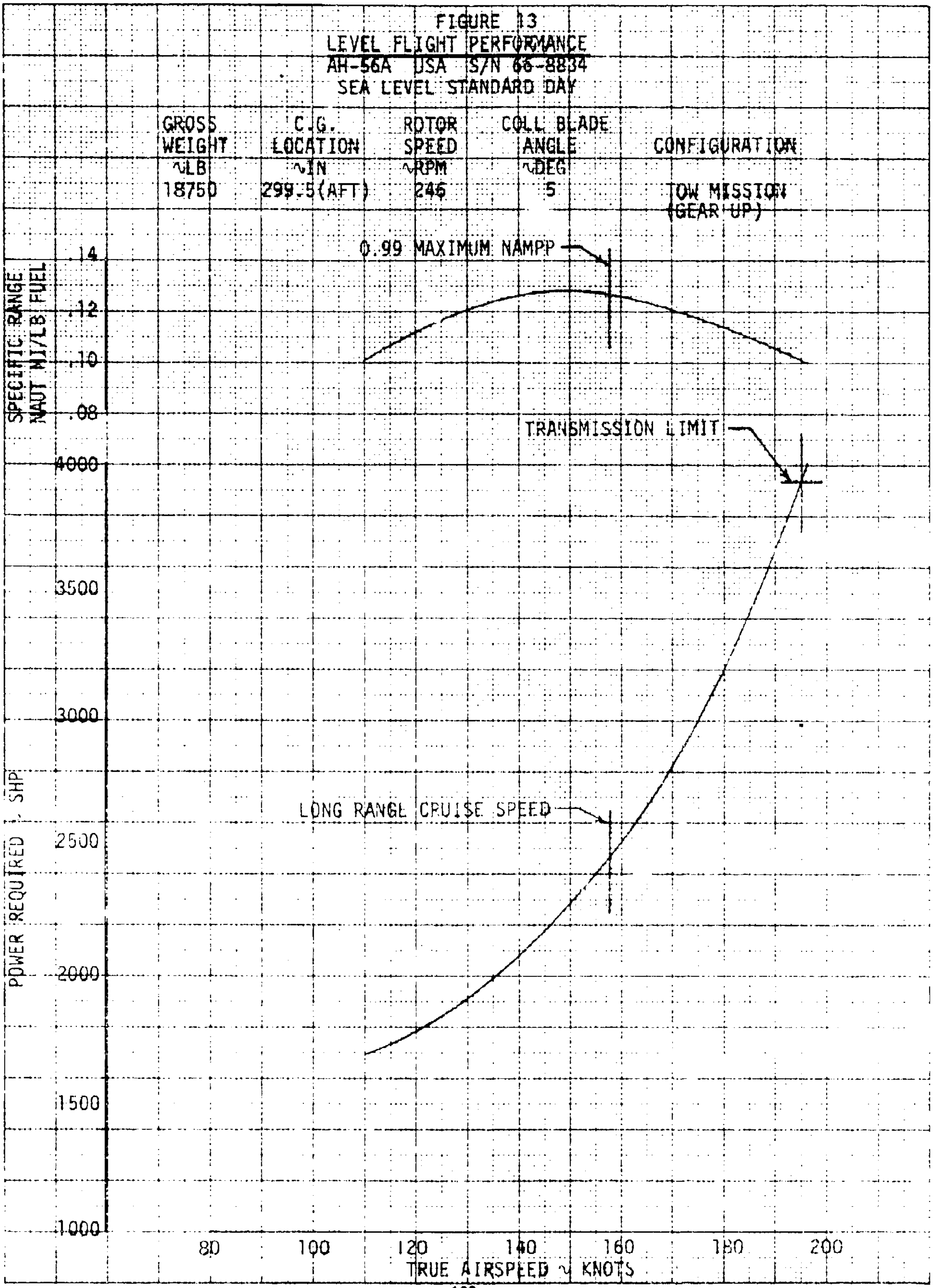
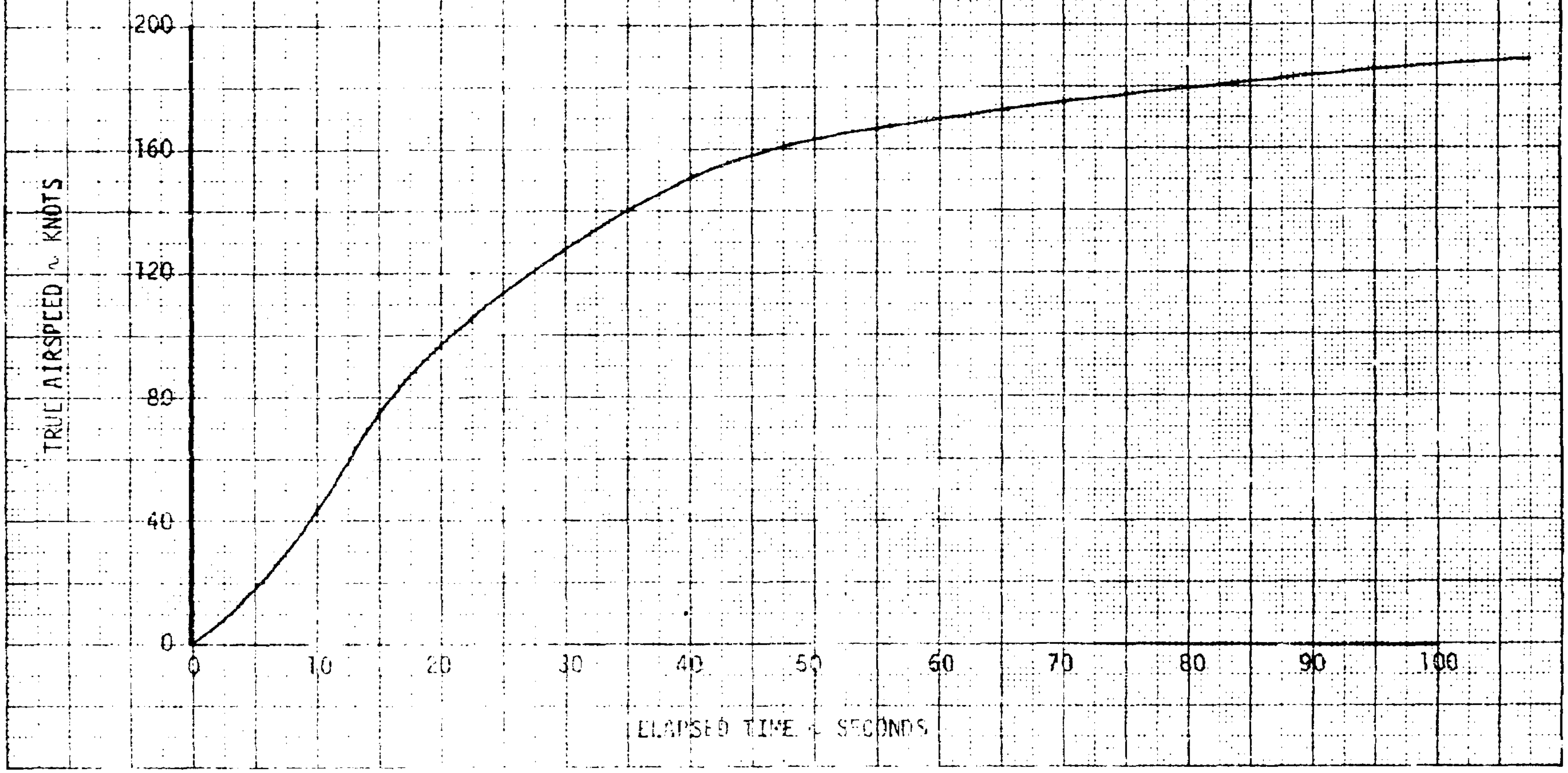


FIGURE 13
LEVEL FLIGHT PERFORMANCE
AH-56A USA S/N 66-8834
SEA LEVEL STANDARD DAY

FIGURE 14
ACCELERATION
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG ROTOR SPEED ~RPM	CONFIGURATION
19210	299.5(AFT)	1310	246	EXT STORES



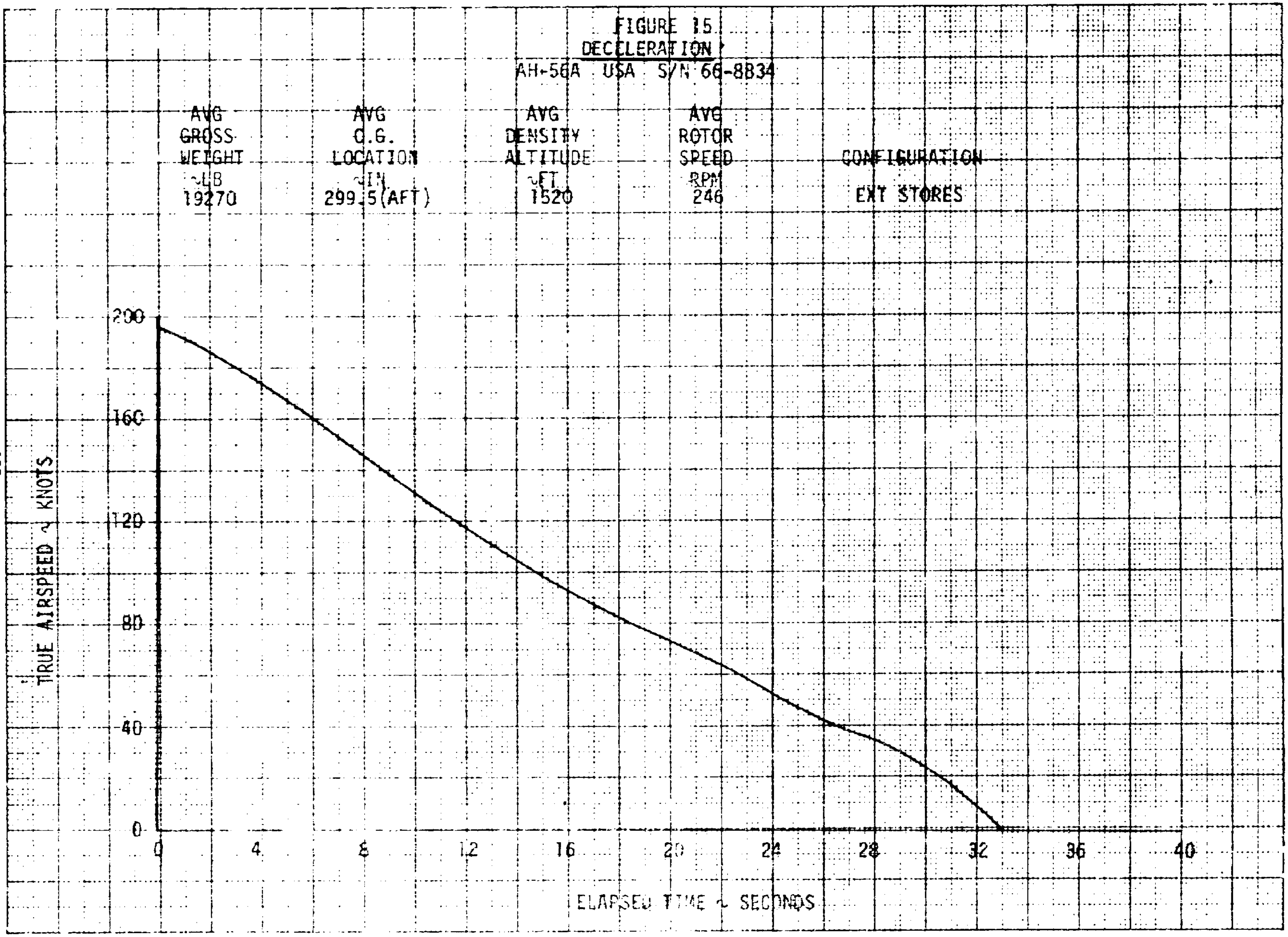


FIGURE 15
DECELERATION
AH-56A USA S/N 66-8834

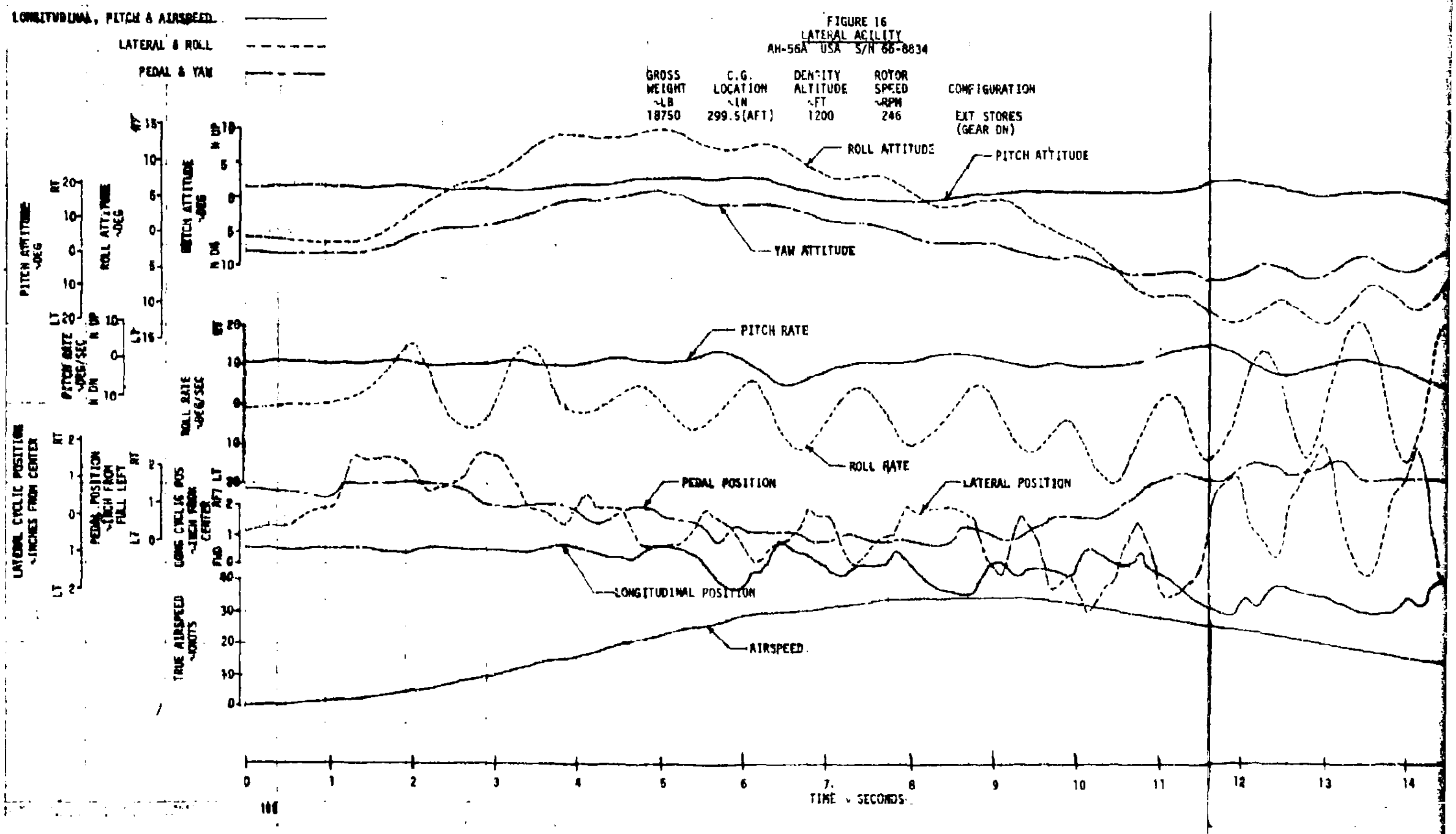
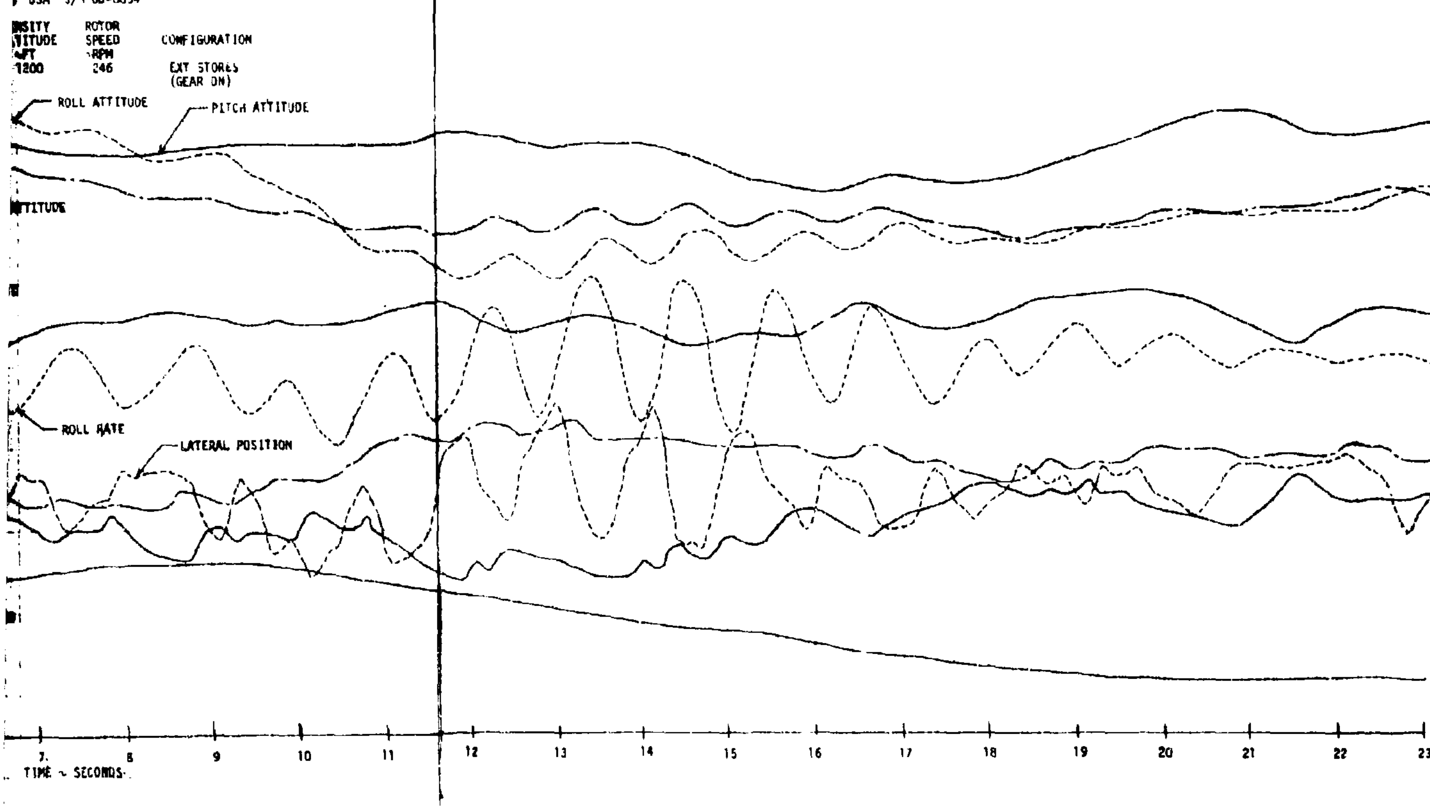


FIGURE 16
LATERAL AGILITY
AH-56A USA S/N 66-8834

FIGURE 16
LATERAL AGILITY
USA S/N 66-8834



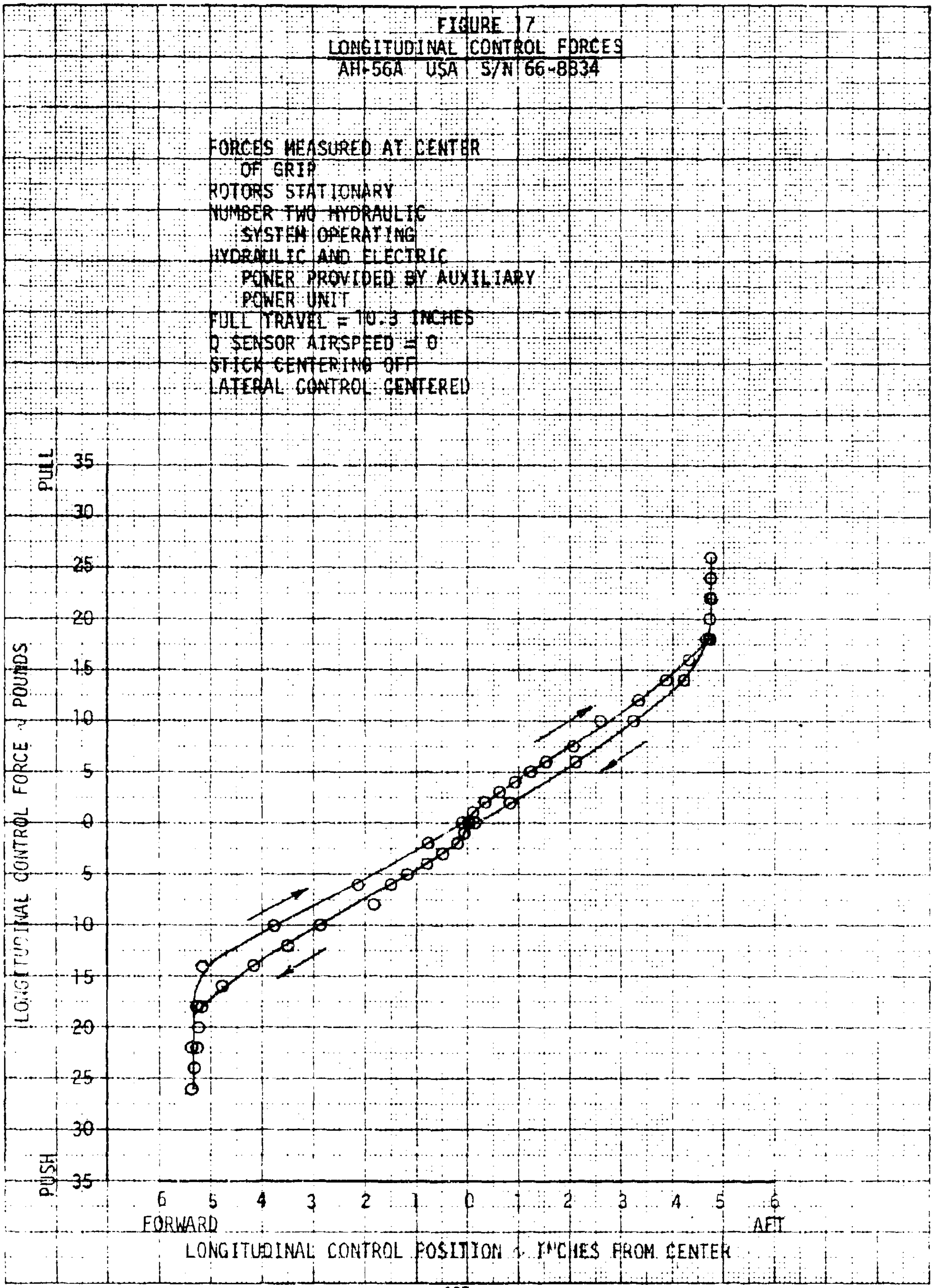


FIGURE 17
LONGITUDINAL CONTROL FORCES
AH-56A USA S/N 66-8834
FORCES MEASURED AT CENTER OF GRIP
ROTORS STATIONARY
NUMBER TWO HYDRAULIC SYSTEM OPERATING
HYDRAULIC AND ELECTRIC POWER PROVIDED BY AUXILIARY POWER UNIT
FULL TRAVEL = 10.3 INCHES
Q SENSOR AIRSPEED = 0
STICK CENTERING OFF
LATERAL CONTROL CENTERED
PULL
PUSH
LONGITUDINAL CONTROL FORCE ~ POUNDS
35
30
25
20
15
10
5
0
5
10
15
20
25
30
35
6 5 4 3 2 1 0 1 2 3 4 5 6
FORWARD
AFT
LONGITUDINAL CONTROL POSITION ~ INCHES FROM CENTER

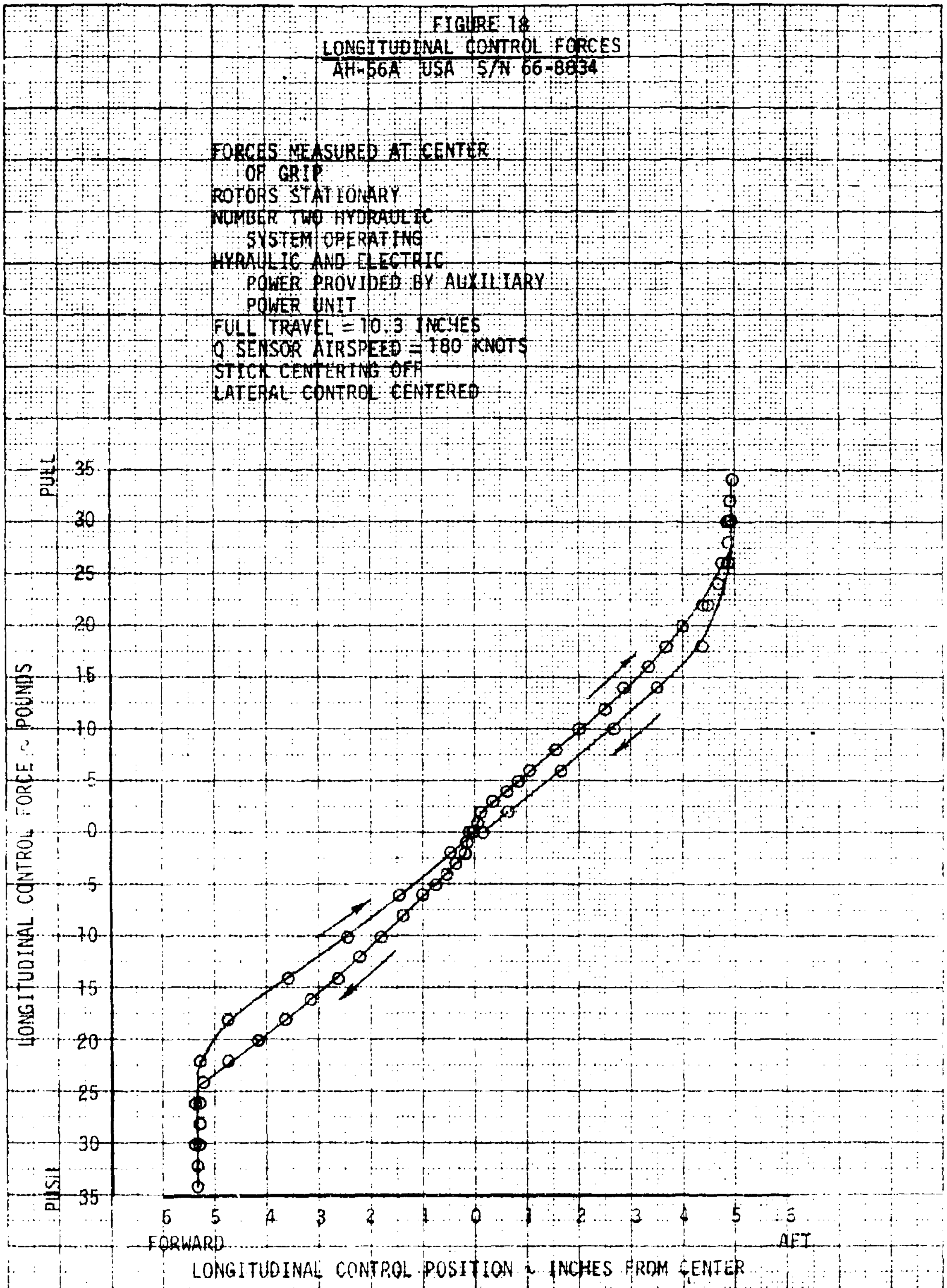


FIGURE 18
LONGITUDINAL CONTROL FORCES
AH-56A USA S/N 66-8834
FORCES MEASURED AT CENTER OF GRIP
ROTORS STATIONARY
NUMBER TWO HYDRAULIC SYSTEM OPERATING
HYRAULIC AND ELECTRIC POWER PROVIDED BY AUXILIARY POWER UNIT
FULL TRAVEL = 10.3 INCHES
Q SENSOR AIRSPEED = 180 KNOTS
STICK CENTERING OFF
LATERAL CONTROL CENTERED
PULL
PUSH
35
30
25
20
15
10
5
0
5
10
15
20
25
30
35
LONGITUDINAL CONTROL FORCE ~ POUNDS
6
5
4
3
2
1
0
1
2
3
4
5
6
FORWARD
AFT
LONGITUDINAL CONTROL POSITION ~ INCHES FROM CENTER

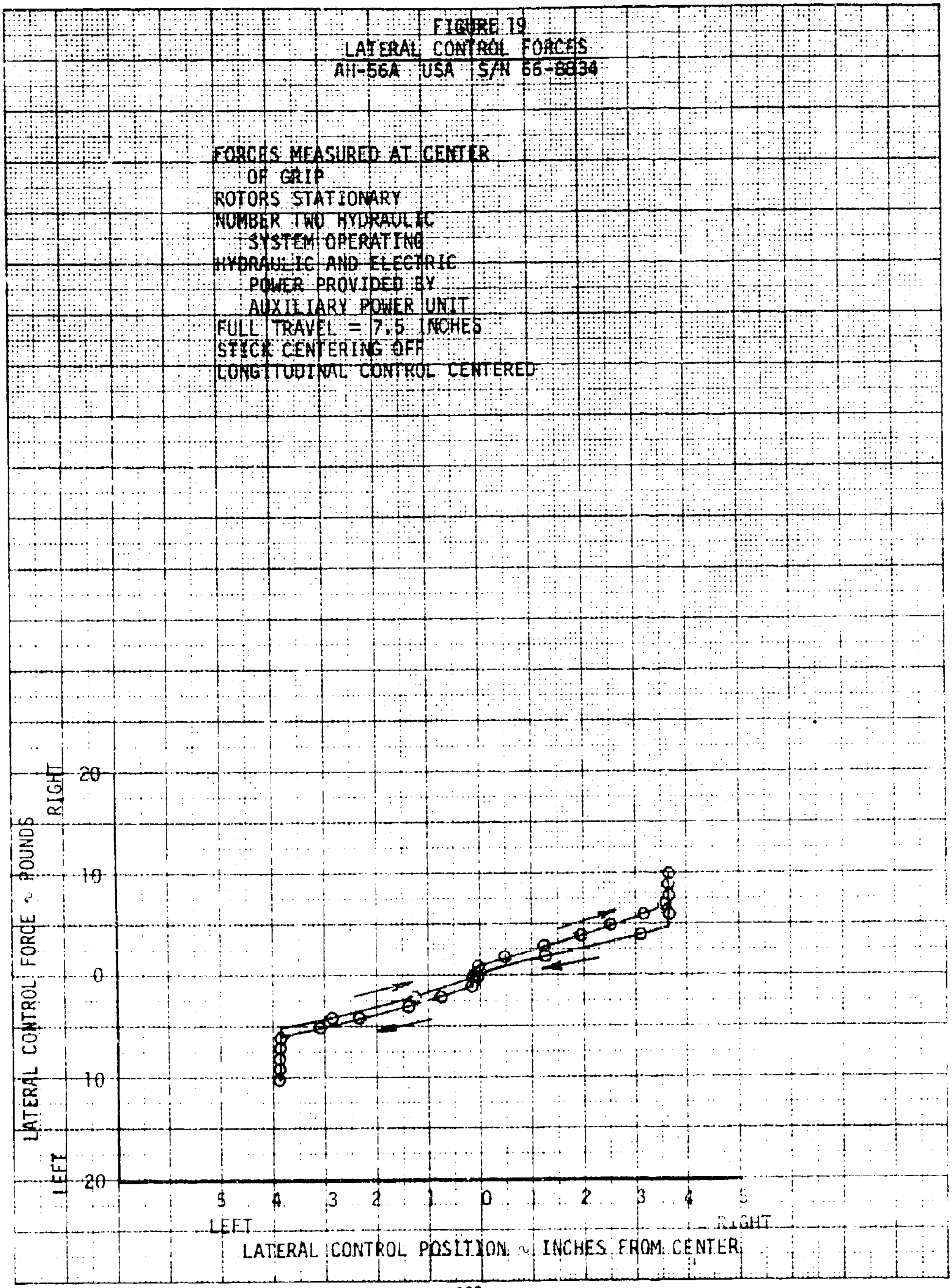


FIGURE 19
LATERAL CONTROL FORCES
AH-56A USA S/N 66-8834
FORCES MEASURED AT CENTER
OF GRIP
ROTORS STATIONARY
NUMBER TWO HYDRAULIC
SYSTEM OPERATING
HYDRAULIC AND ELECTRIC
POWER PROVIDED BY
AUXILIARY POWER UNIT
FULL TRAVEL = 7.5 INCHES
STICK CENTERING OFF
LONGITUDINAL CONTROL CENTERED
LATERAL CONTROL FORCE ~ POUNDS
RIGHT
LEFT
20
10
0
10
20
5
4
3
2
1
0
1
2
3
4
5
LEFT
RIGHT
LATERAL CONTROL POSITION ~ INCHES FROM CENTER

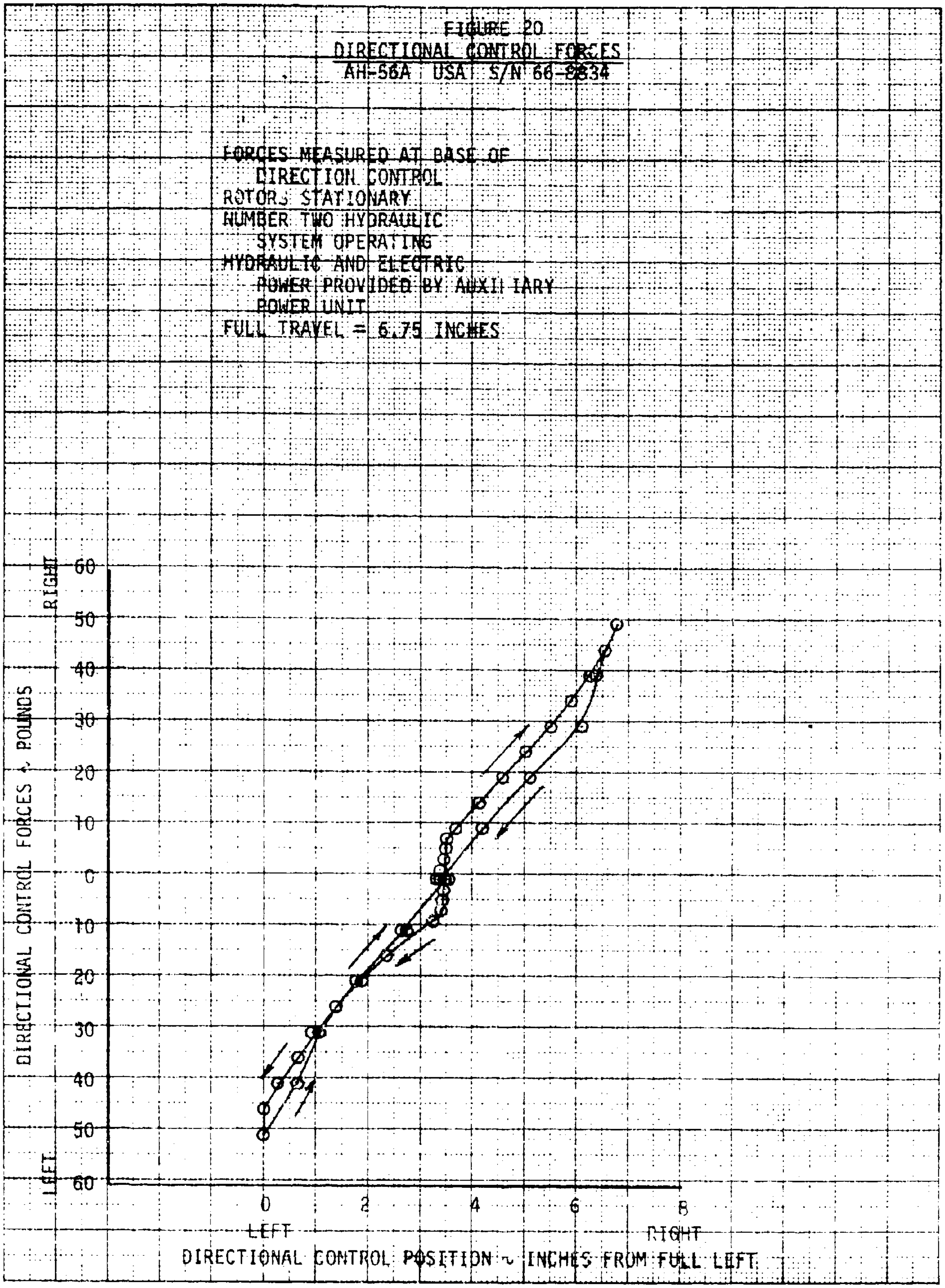


FIGURE 20
DIRECTIONAL CONTROL FORCES
AH-56A USA S/N 66-8834
FORCES MEASURED AT BASE OF
DIRECTION CONTROL
ROTORS STATIONARY
NUMBER TWO HYDRAULIC
SYSTEM OPERATING
HYDRAULIC AND ELECTRIC
POWER PROVIDED BY AUXILIARY
POWER UNIT
FULL TRAVEL = 6.75 INCHES
DIRECTIONAL CONTROL FORCES ~ POUNDS
RIGHT
LEFT
60
50
40
30
20
10
0
10
20
30
40
50
60
0
2
4
6
8
LEFT
RIGHT
DIRECTIONAL CONTROL POSITION ~ INCHES FROM FULL LEFT

FIGURE 21
LIMITS OF CYCLIC CONTROL TRAVEL
AH-56A USA S/N 66-8834

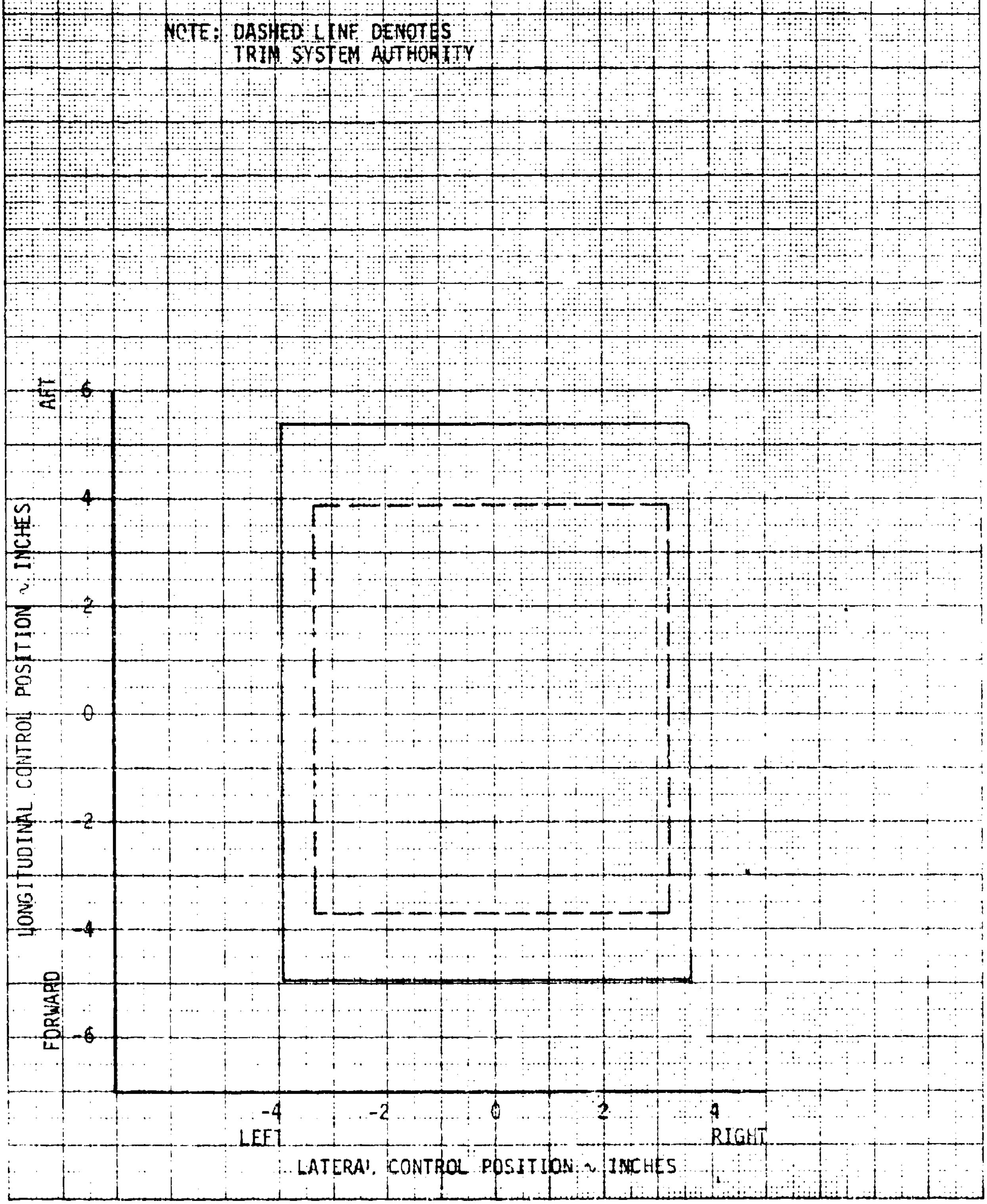


FIGURE 22
TRIM SHIFTS DURING TRANSITION
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	CONFIGURATION
19500	298.8 (AFT)	1100	246	EXT STORES (GEAR DOWN)

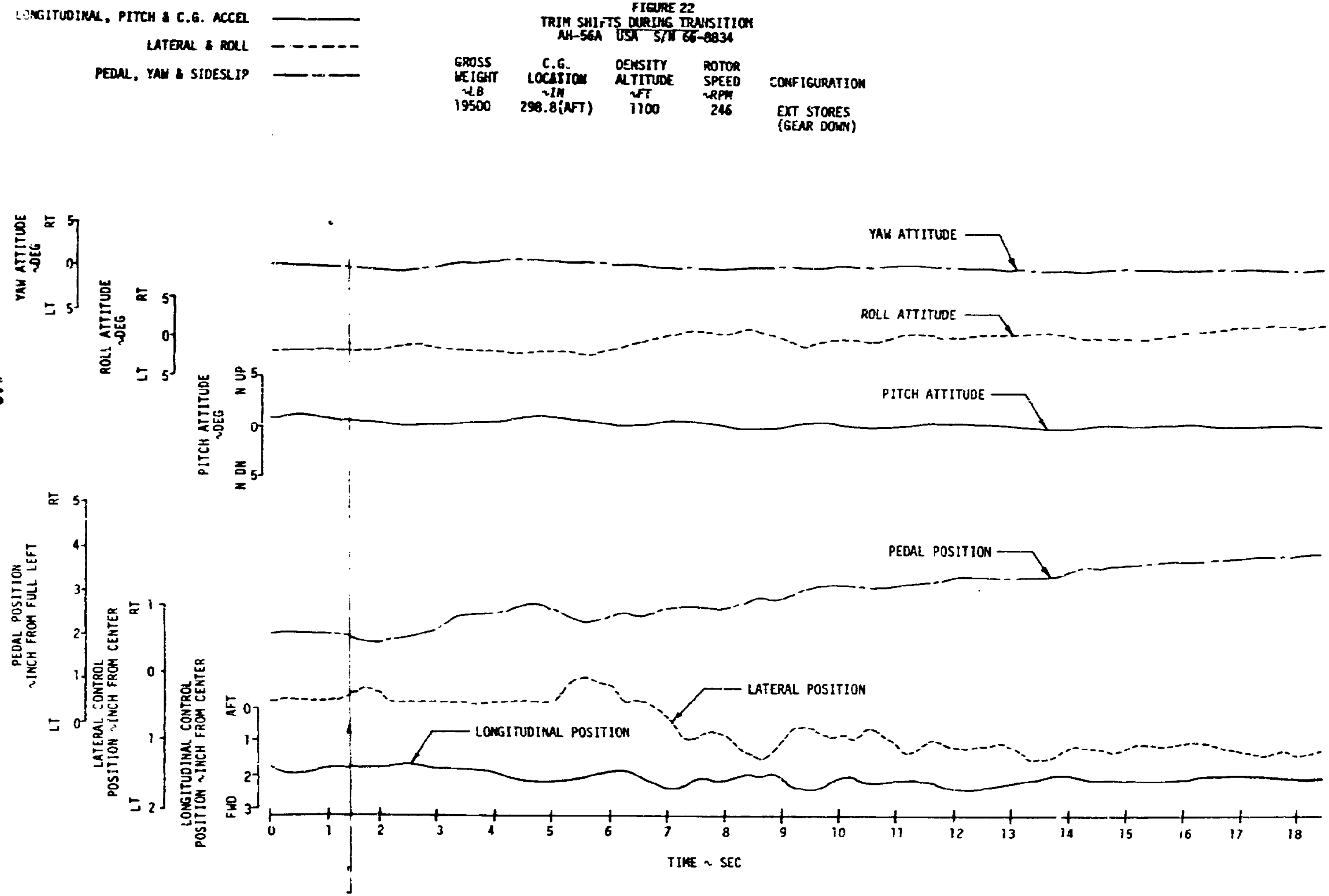


FIGURE 23
CONTROL POSITIONS IN SIDEWARD FLIGHT
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG ROTOR SPEED ~RPM	AVG WHEEL HEIGHT ~FT	PROP BLADE ANGLE ~DEG	CONFIGURATION
18930	296.5(MID)	1200	246	10	-2.2	EXT. STORES (GEAR DOWN)

NOTE: BARS DENOTE MAXIMUM CONTROL EXCURSIONS

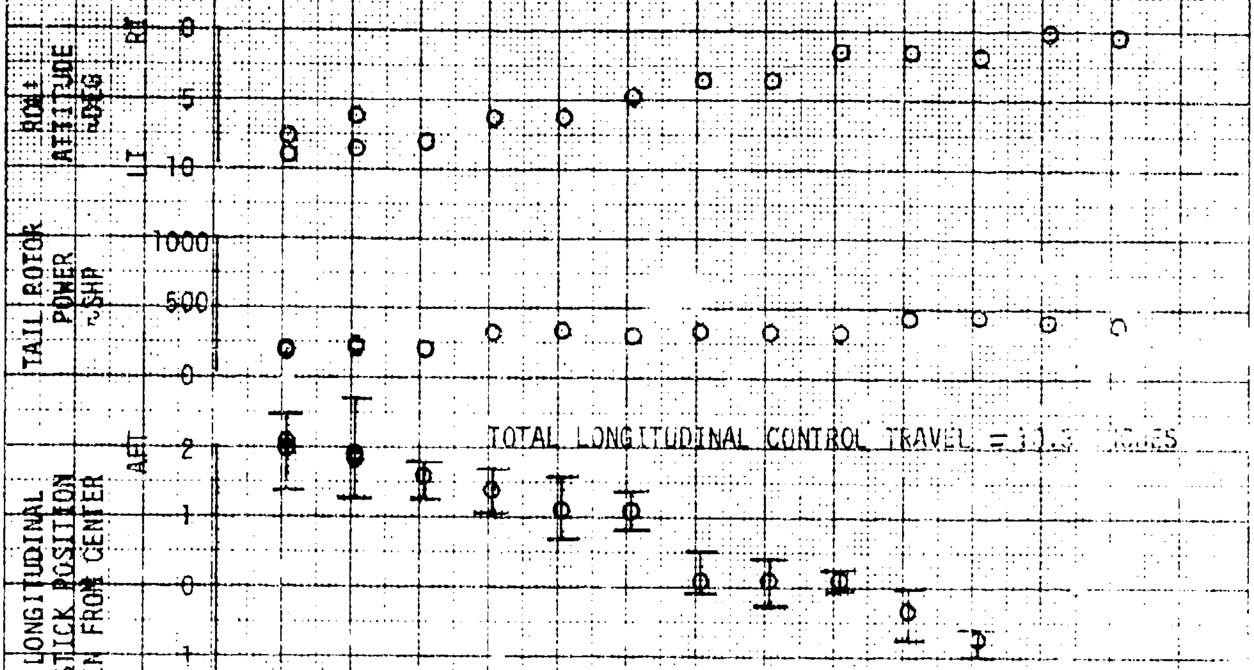
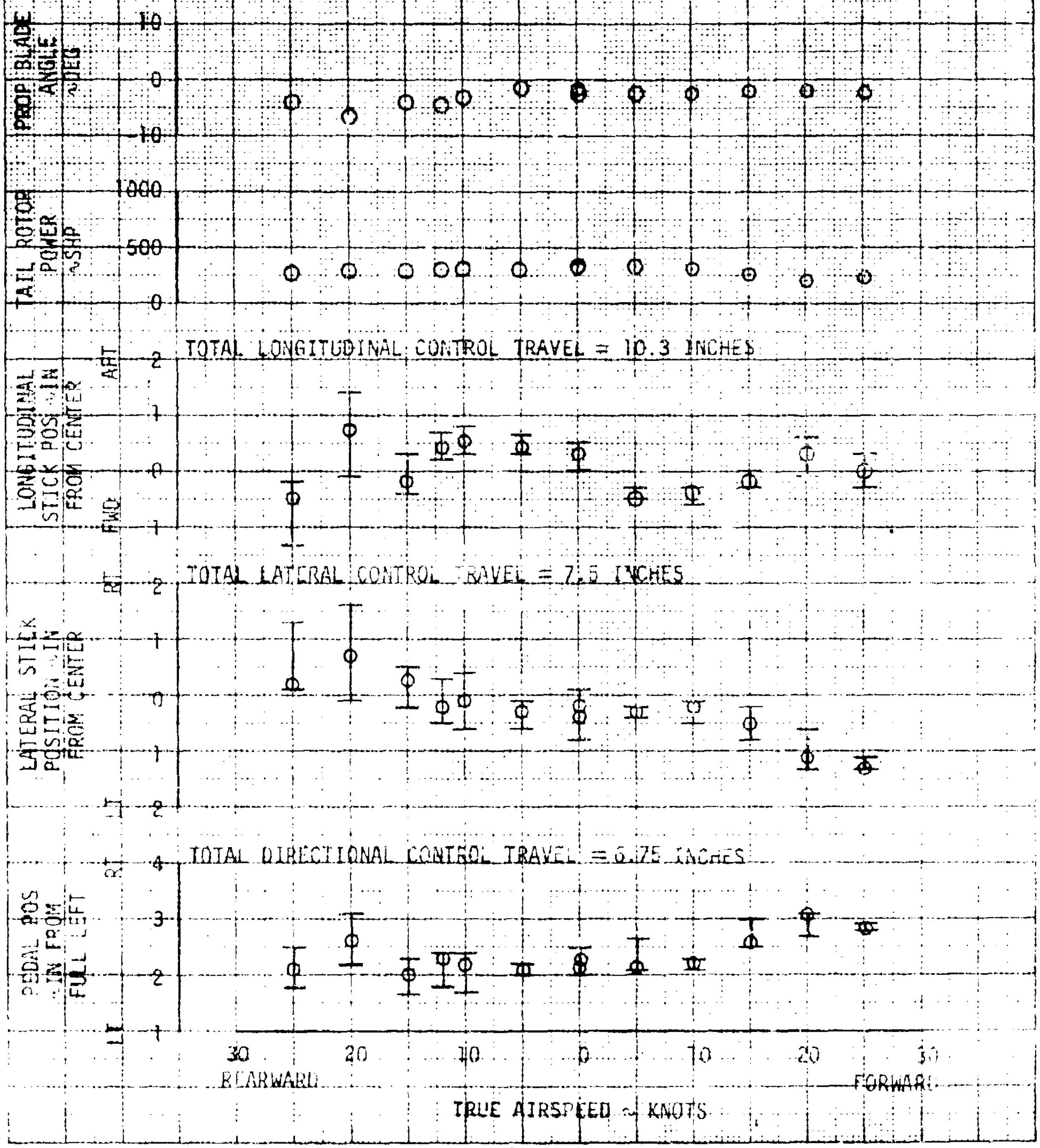


FIGURE 24
CONTROL POSITIONS IN REARWARD FLIGHT
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG ROTOR SPEED ~RPM	AVG WHEEL HEIGHT ~FT	CONFIGURATION
18930	295.5(MID)	1200	246	10	EXT STORES (GEAR DOWN)

NOTE: BARS DENOTE MAXIMUM CONTROL EXCURSIONS



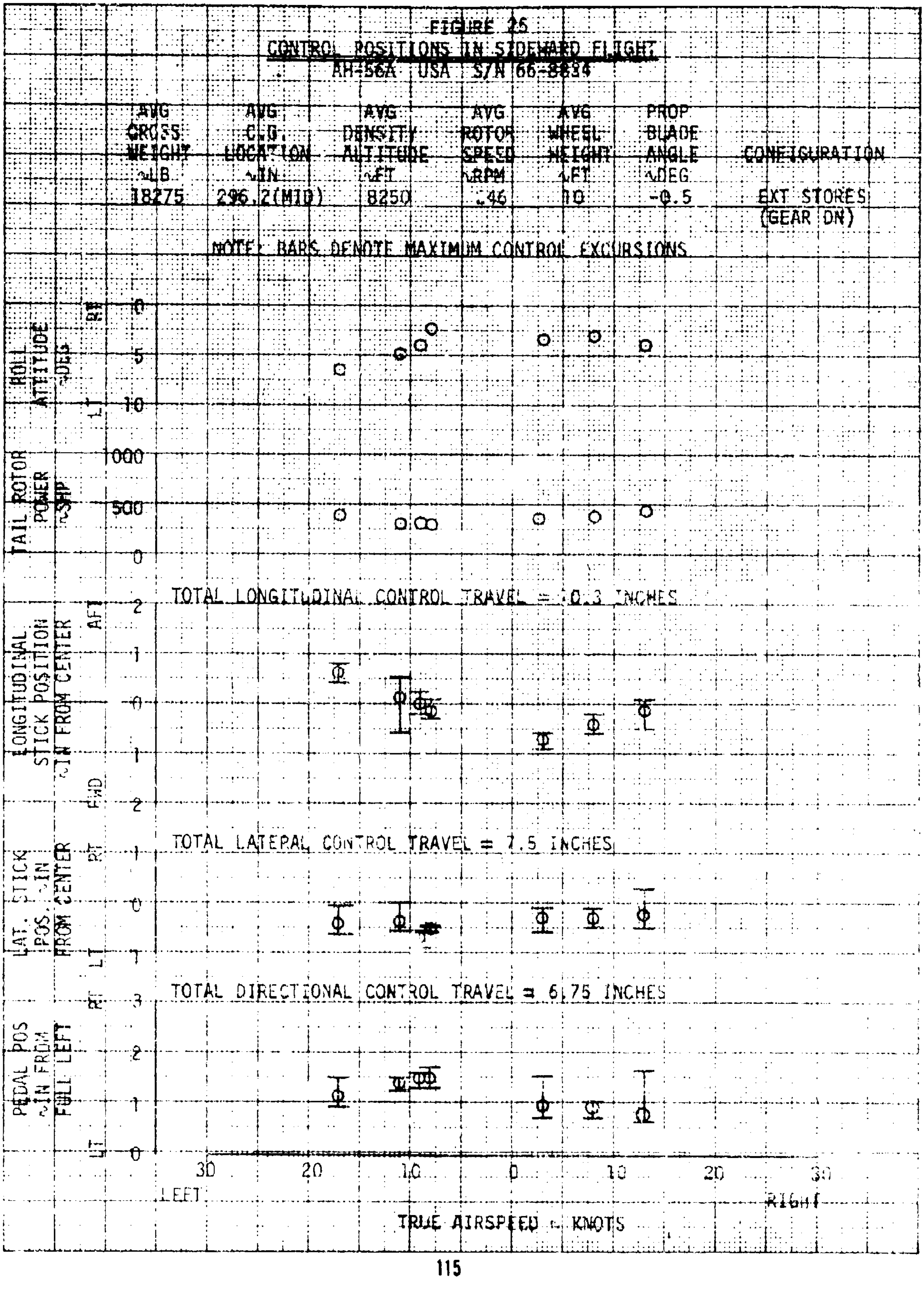
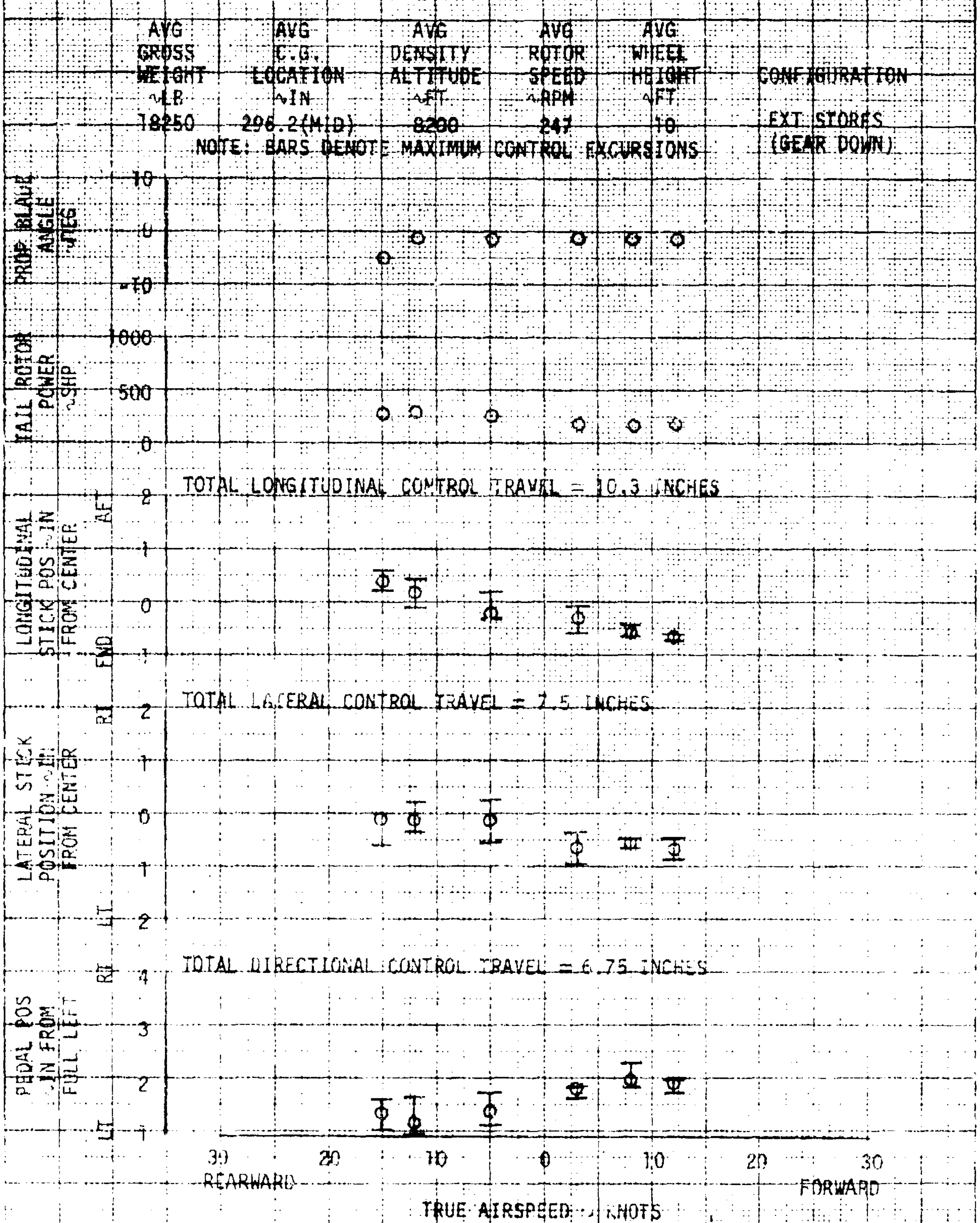


FIGURE 25
CONTROL POSITIONS IN SIDEWARD FLIGHT
AH-56A USA S/N 66-8834

FIGURE 26
CONTROL POSITIONS IN REARWARD FLIGHT
AH-56A USA S/N 66-8834



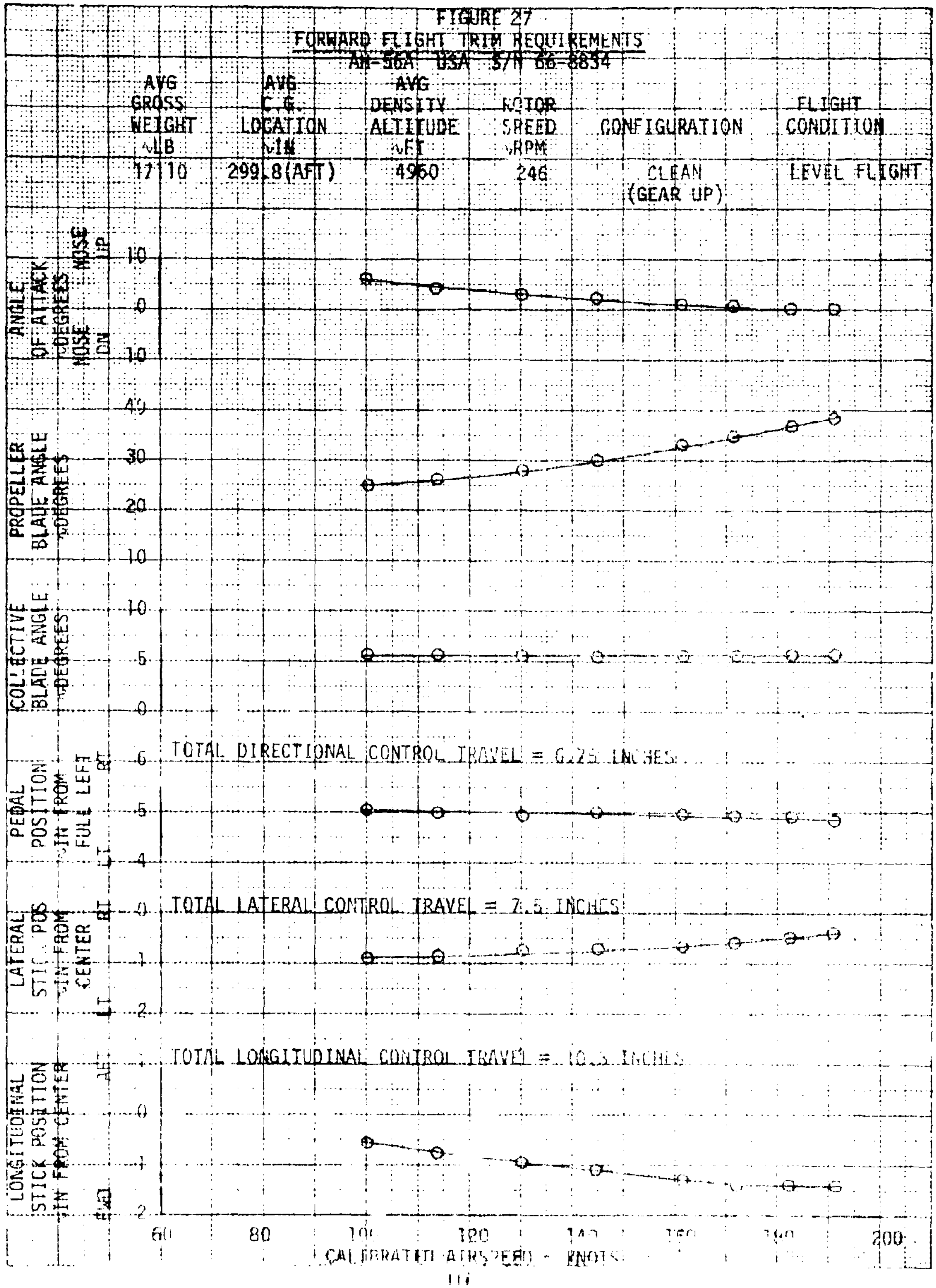


FIGURE 27
FORWARD FLIGHT TRIM REQUIREMENTS
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB
17110
AVG C.G. LOCATION ~IN
299.8(AFT)
AVG DENSITY ALTITUDE ~FT
4960
ROTOR SPEED ~RPM
246
CONFIGURATION
CLEAN (GEAR UP)
FLIGHT CONDITION
LEVEL FLIGHT
ANGLE OF ATTACK ~DEGREES
NOSE UP
NOSE DN
PROPELLER BLADE ANGLE ~DEGREES
COLLECTIVE BLADE ANGLE ~DEGREES
PEDAL POSITION ~IN FROM FULL LEFT
RT
LT
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.25 INCHES
LATERAL STICK POS ~IN FROM CENTER
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LONGITUDINAL STICK POSITION ~IN FROM CENTER
AFT
FWD
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.5 INCHES
CALIBRATED AIRSPEED ~ KNOTS

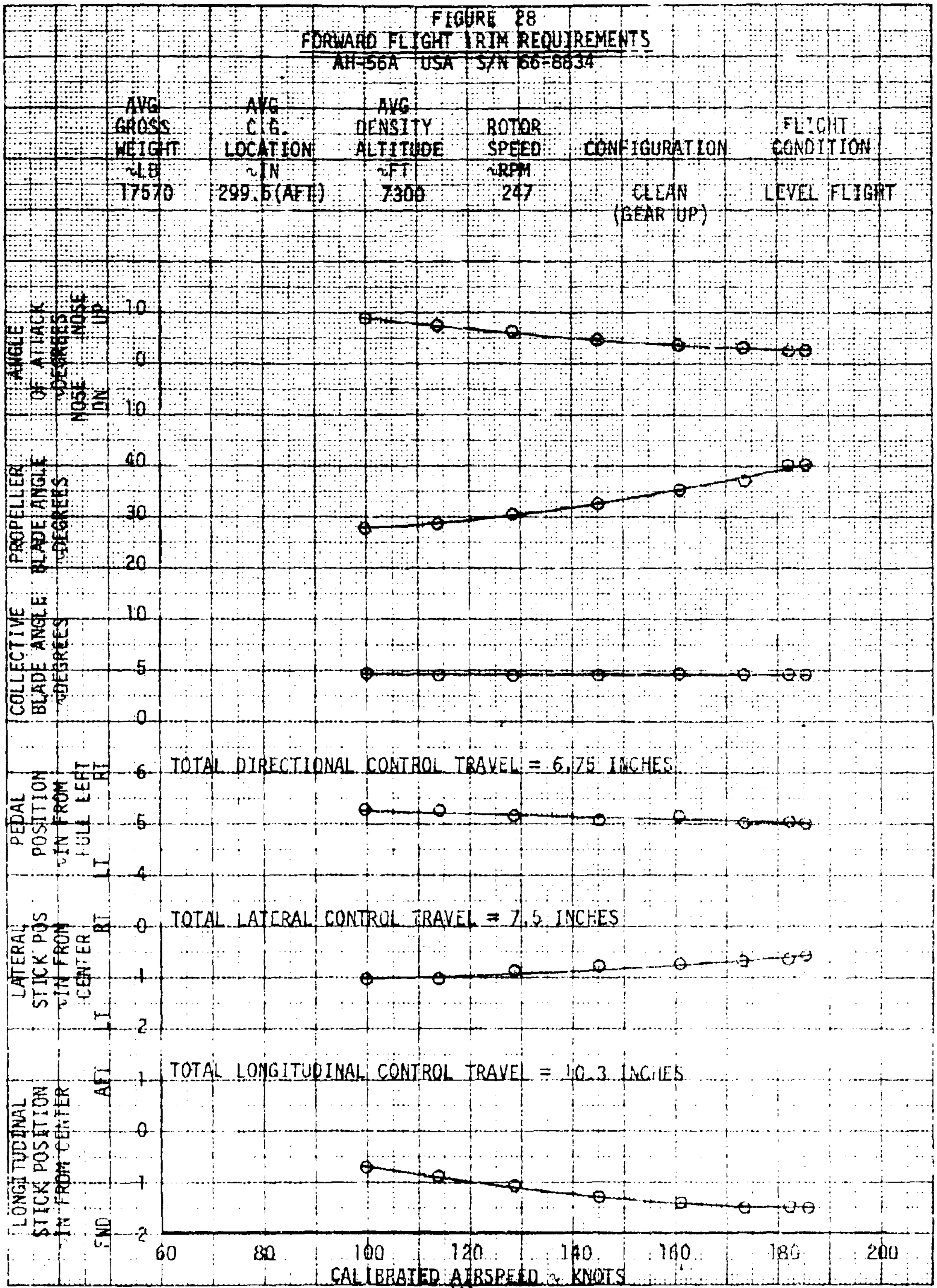


FIGURE 28
FORWARD FLIGHT TRIM REQUIREMENTS
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 17570
AVG C.G. LOCATION ~IN 299.6(AFT)
AVG DENSITY ALTITUDE ~FT 7300
ROTOR SPEED ~RPM 247
CONFIGURATION CLEAN (GEAR UP)
FLIGHT CONDITION LEVEL FLIGHT
ANGLE OF ATTACK ~DEGREES NOSE UP NOSE DN
PROPELLER BLADE ANGLE ~DEGREES
COLLECTIVE BLADE ANGLE ~DEGREES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75 INCHES
PEDAL POSITION ~IN FROM FULL LEFT RT LT
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LATERAL STICK POS ~IN FROM CENTER RT LT
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
LONGITUDINAL STICK POSITION ~IN FROM CENTER AFT FWD
CALIBRATED AIRSPEED ~ KNOTS

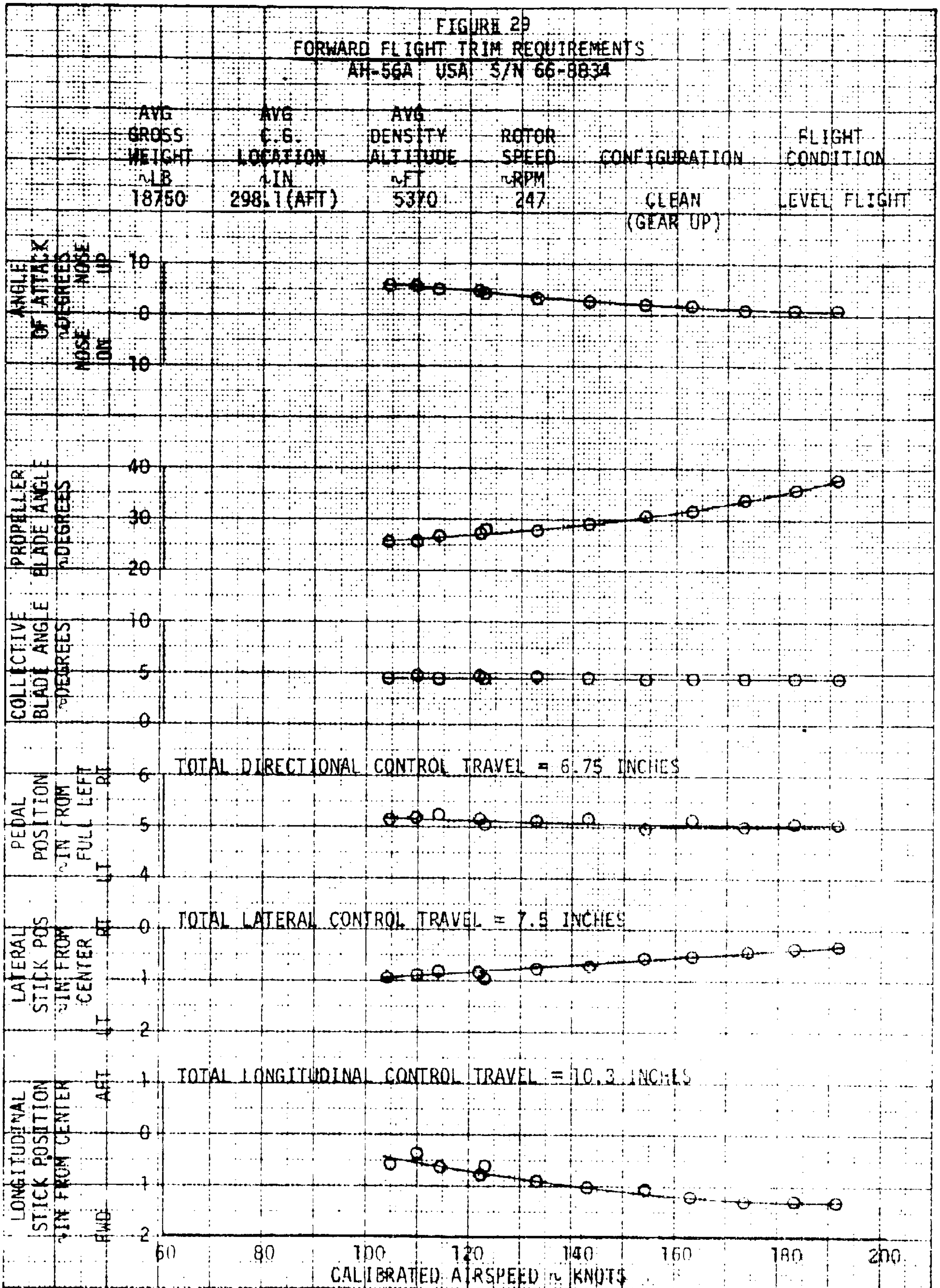
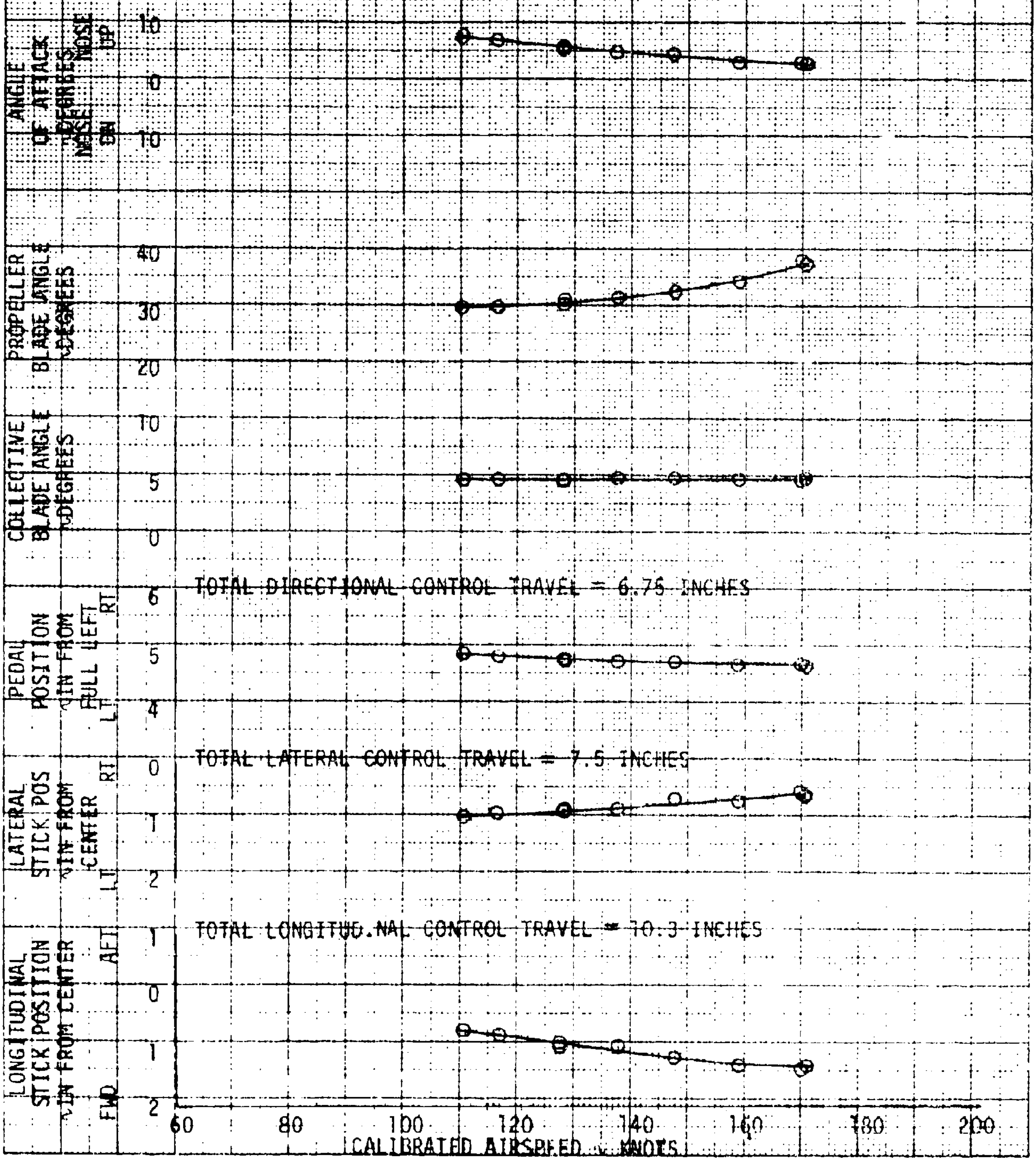


FIGURE 29
FORWARD FLIGHT TRIM REQUIREMENTS
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 18750
AVG C.G. LOCATION ~IN 298.1(AFT)
AVG DENSITY ALTITUDE ~FT 5370
ROTOR SPEED ~RPM 247
CONFIGURATION CLEAN (GEAR UP)
FLIGHT CONDITION LEVEL FLIGHT
ANGLE OF ATTACK ~DEGREES
NOSE UP
NOSE DN
PROPELLER BLADE ANGLE ~DEGREES
COLLECTIVE BLADE ANGLE ~DEGREES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75 INCHES
PEDAL POSITION ~IN FROM FULL LEFT
RT
LT
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LATERAL STICK POS ~IN FROM CENTER
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
LONGITUDINAL STICK POSITION ~IN FROM CENTER
AFT
FWD
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 30
FORWARD FLIGHT TRIM REQUIREMENTS
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
18490	298.3(AFT)	8270	246	CLEAN (GEAR UP)	LEVEL FLIGHT



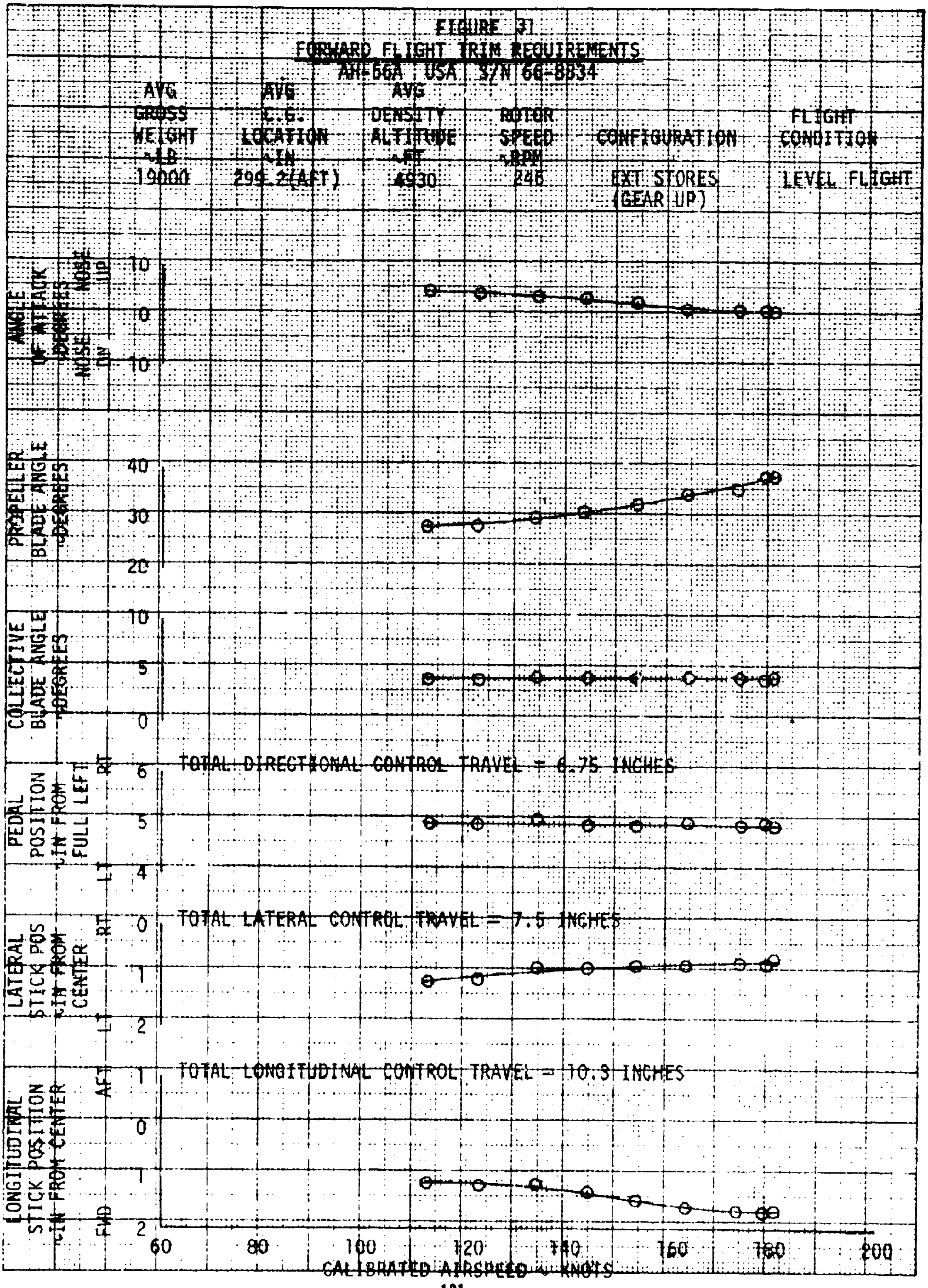
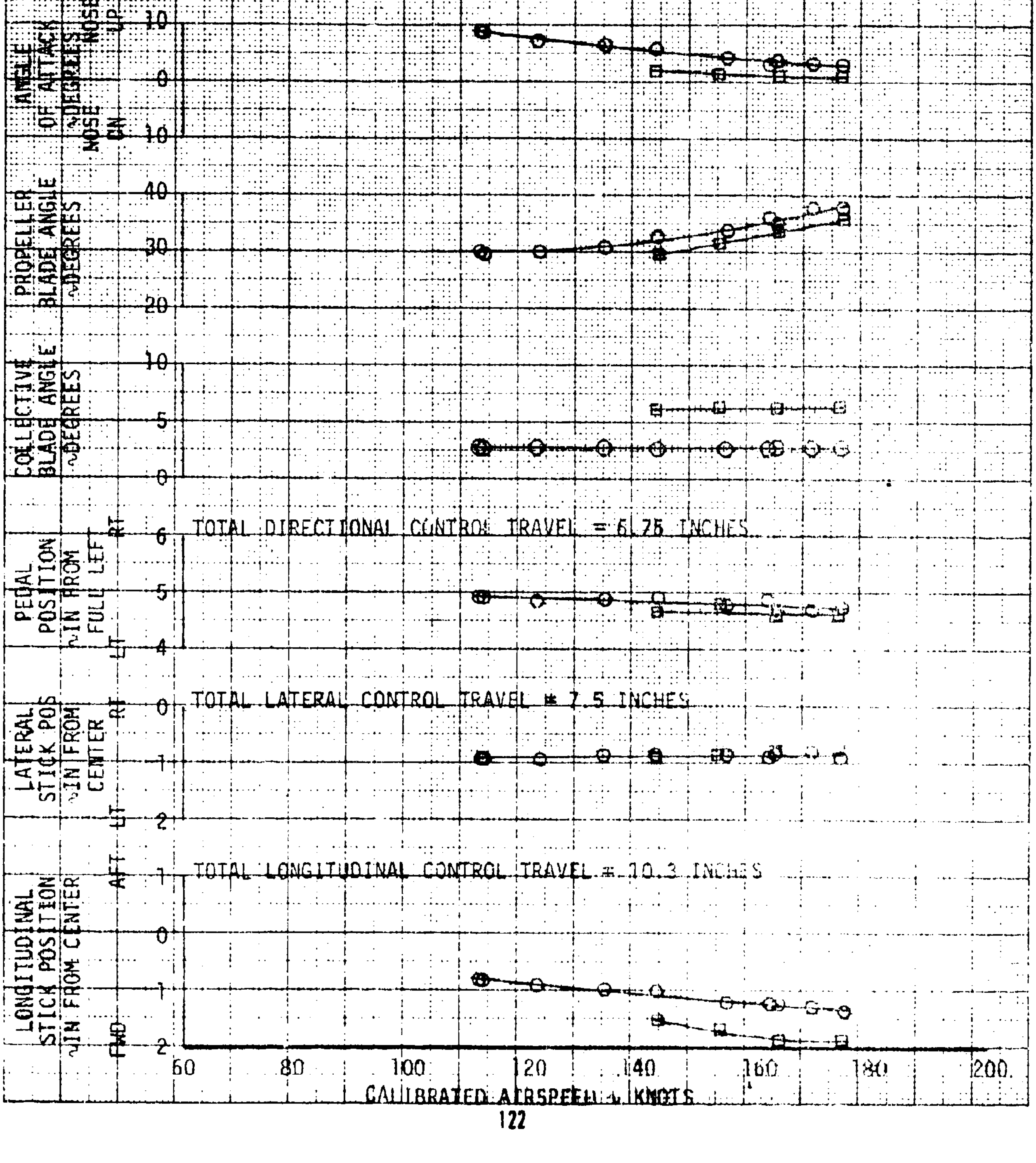


FIGURE 31
FORWARD FLIGHT TRIM REQUIREMENTS
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB
19000
AVG C.G. LOCATION ~IN
299.2(AFT)
AVG DENSITY ALTITUDE ~FT
4930
ROTOR SPEED ~RPM
246
CONFIGURATION
EXT STORES (GEAR UP)
FLIGHT CONDITION
LEVEL FLIGHT
ANGLE OF ATTACK ~DEGREES
NOSE UP
NOSE DN
PROPELLER BLADE ANGLE ~DEGREES
COLLECTIVE BLADE ANGLE ~DEGREES
PEDAL POSITION ~IN FROM FULL LEFT
RT
LT
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75 INCHES
LATERAL STICK POS ~IN FROM CENTER
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LONGITUDINAL STICK POSITION ~IN FROM CENTER
AFT
FWD
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

FIGURE 32
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
○	18930	299.1(AFT)	4490	248	EXT STORES (GEAR UP)	LEVEL FLIGHT
□	18130	299.5(AFT)	7190	249	EXT STORES (GEAR UP)	LEVEL FLIGHT



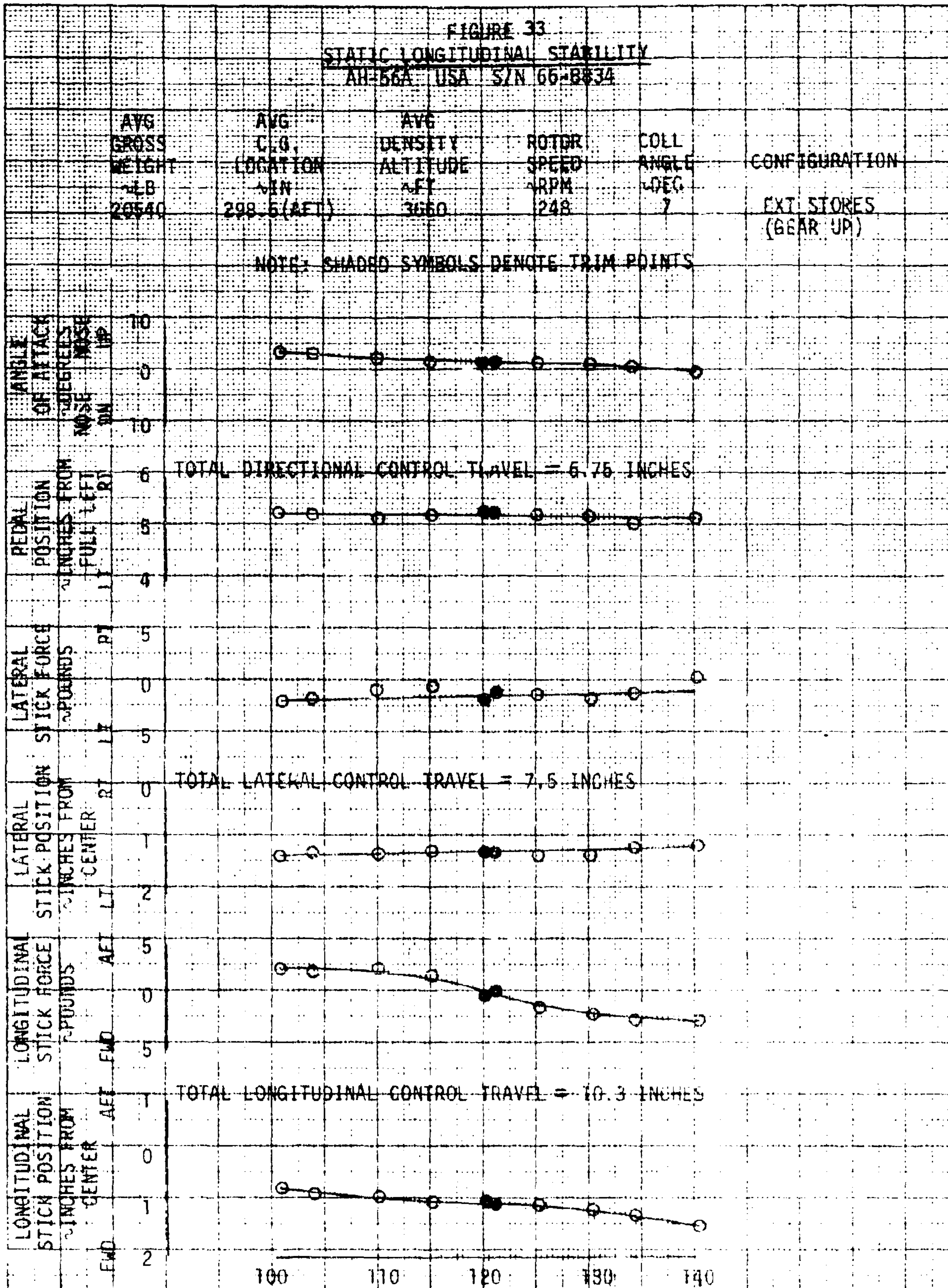


FIGURE 33
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 20540
AVG C.G. LOCATION ~IN 298.5(AFT)
AVG DENSITY ALTITUDE ~FT 3660
ROTOR SPEED ~RPM 248
COLL ANGLE ~DEG 7
CONFIGURATION EXT STORES (GEAR UP)
NOTE: SHADED SYMBOLS DENOTE TRIM POINTS
ANGLE OF ATTACK ~DEGREES NOSE UP NOSE DN
PEDAL POSITION ~INCHES FROM FULL LEFT RT LT
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75 INCHES
LATERAL STICK FORCE ~POUNDS RT LT
LATERAL STICK POSITION ~INCHES FROM CENTER RT LT
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LONGITUDINAL STICK FORCE ~POUNDS AFT FWD
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER AFT FWD
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
CALIBRATED AIRSPEED ~ KNOTS
100
110
120
130
140

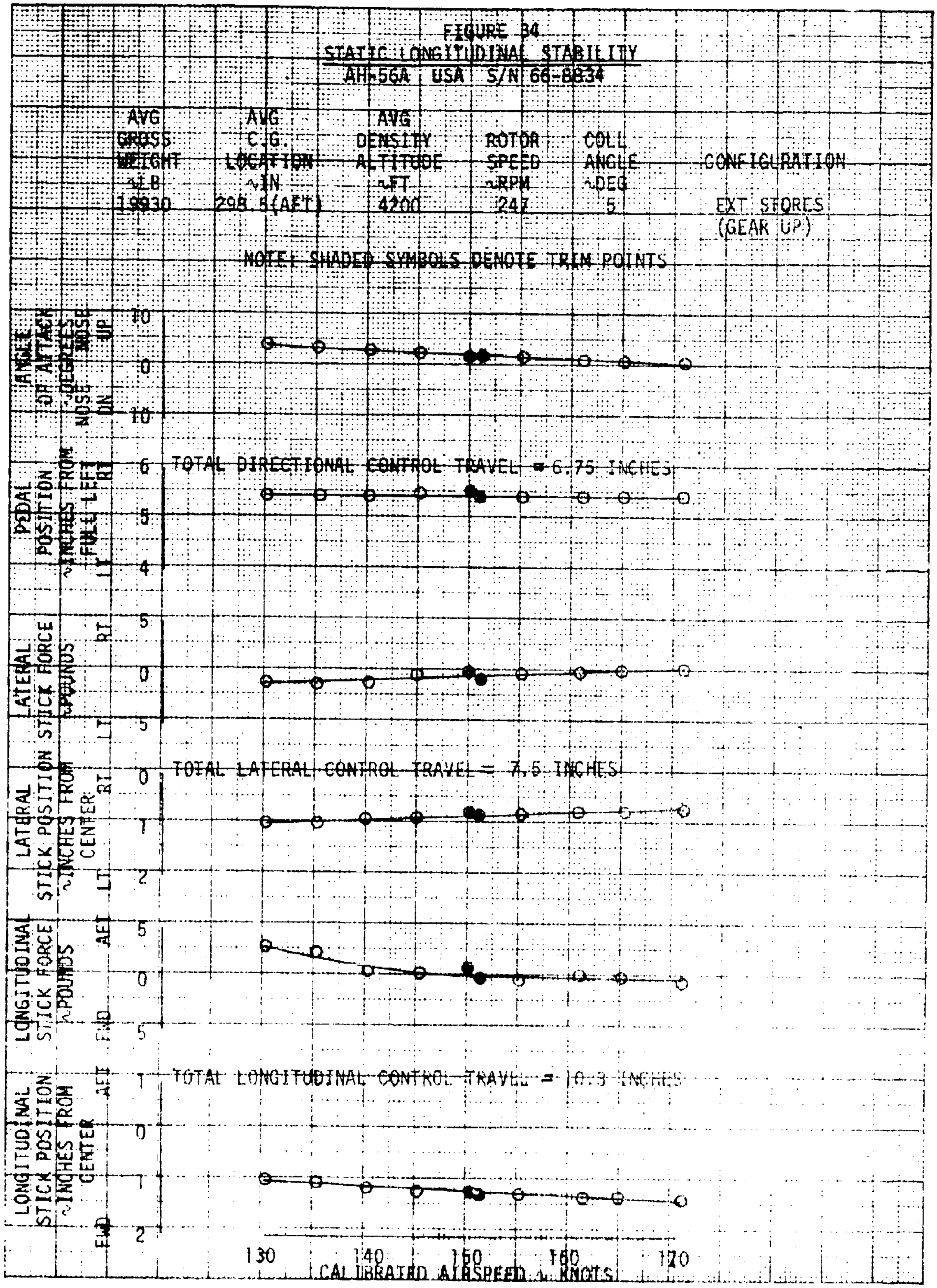


FIGURE 34
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB
19930
AVG C.G. LOCATION ~IN
298.5(AFT)
AVG DENSITY ALTITUDE ~FT
4200
ROTOR SPEED ~RPM
247
COLL ANGLE ~DEG
5
CONFIGURATION
EXT STORES (GEAR UP)
NOTE: SHADED SYMBOLS DENOTE TRIM POINTS
ANGLE OF ATTACK ~DEGREES NOSE DN NOSE UP
PEDAL POSITION ~INCHES FROM FULL LEFT LT RT
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75 INCHES
LATERAL STICK FORCE ~POUNDS LT RT
LATERAL STICK POSITION ~INCHES FROM CENTER LT RT
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LONGITUDINAL STICK FORCE ~POUNDS FWD AFT
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER FWD AFT
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
CALIBRATED AIRSPEED ~ KNOTS
130
140
150
160
170

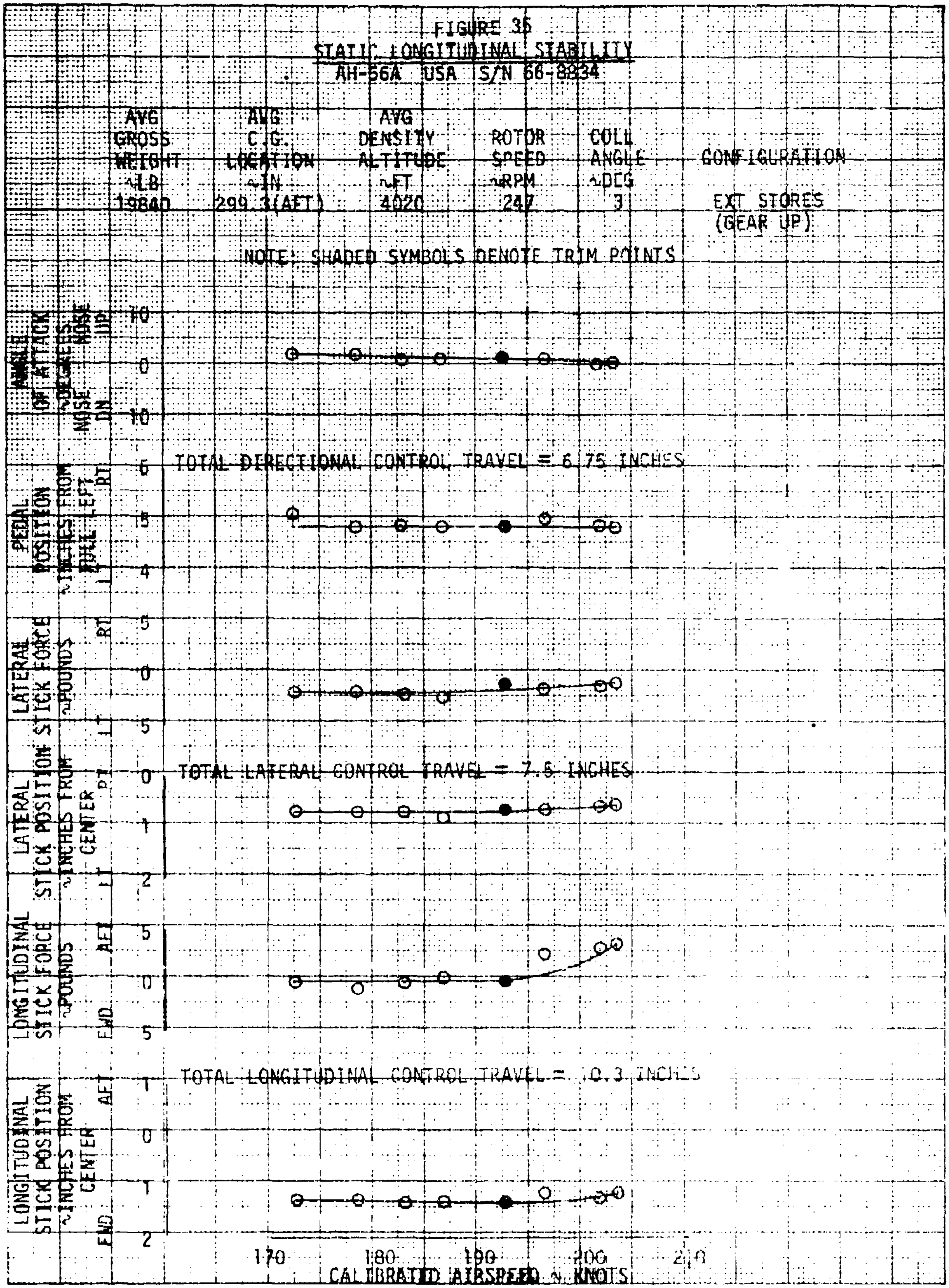


FIGURE 35
STATIC LONGITUDINAL STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 19840
AVG C.G. LOCATION ~IN 299.3(AFT)
AVG DENSITY ALTITUDE ~FT 4020
ROTOR SPEED ~RPM 247
COLL ANGLE ~DEG 3
CONFIGURATION EXT STORES (GEAR UP)
NOTE: SHADED SYMBOLS DENOTE TRIM POINTS
ANGLE OF ATTACK ~DEGREES NOSE UP NOSE DN
PEDAL POSITION ~INCHES FROM FULL LEFT LT RT
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75 INCHES
LATERAL STICK FORCE ~POUNDS LT RT
LATERAL STICK POSITION ~INCHES FROM CENTER LT RT
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LONGITUDINAL STICK FORCE ~POUNDS FWD AFT
LONGITUDINAL STICK POSITION ~INCHES FROM CENTER FWD AFT
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
CALIBRATED AIRSPEED ~ KNOTS
170 180 190 200 210

FIGURE 36

STATIC LATERAL-DIRECTIONAL STABILITY

AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
20150	299.3(AFT)	5100	250	6.9	118	EXT STORES (GEAR UP)

NOTE: SHADED SYMBOLS DENOTE TRIM POINTS

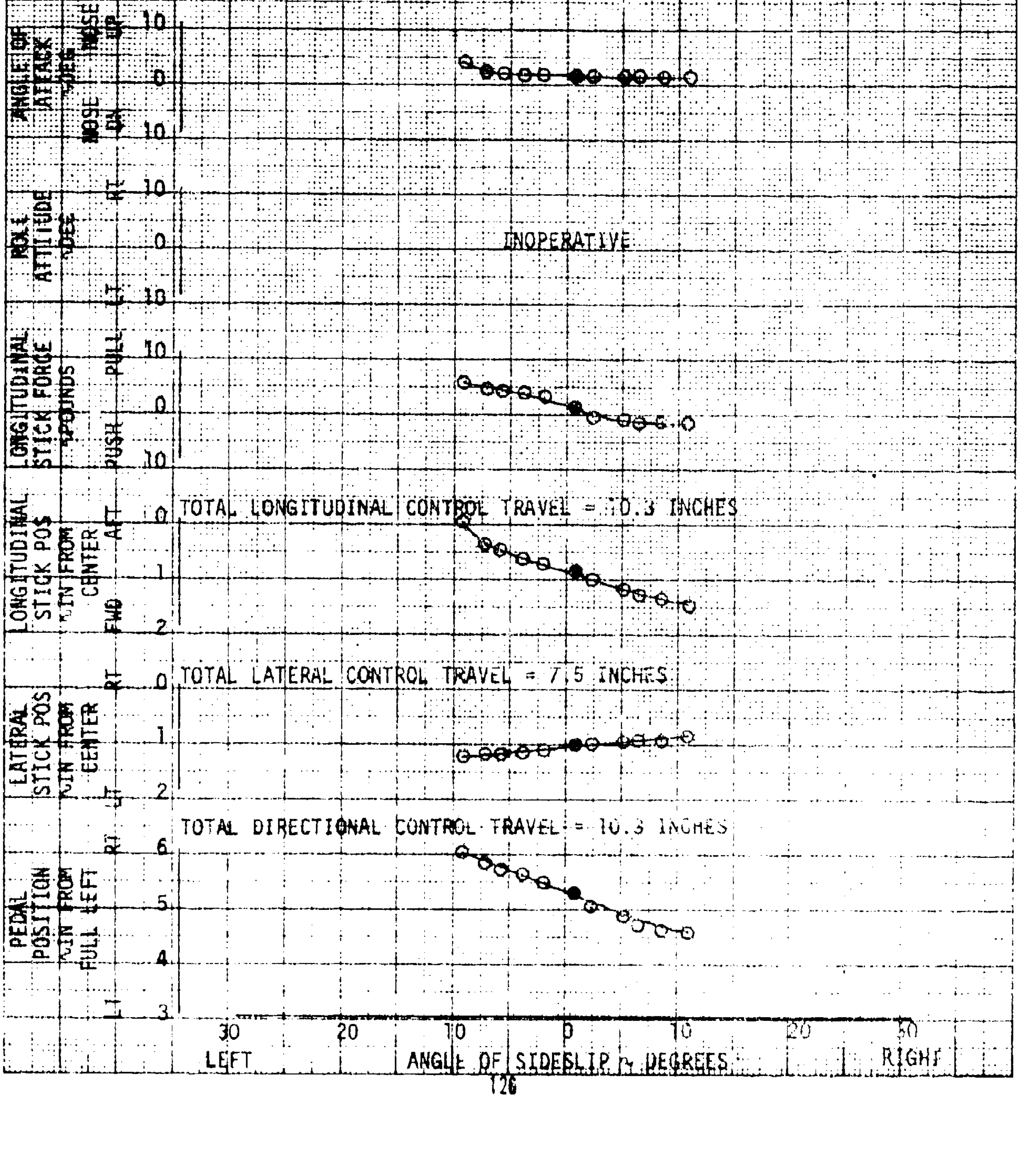
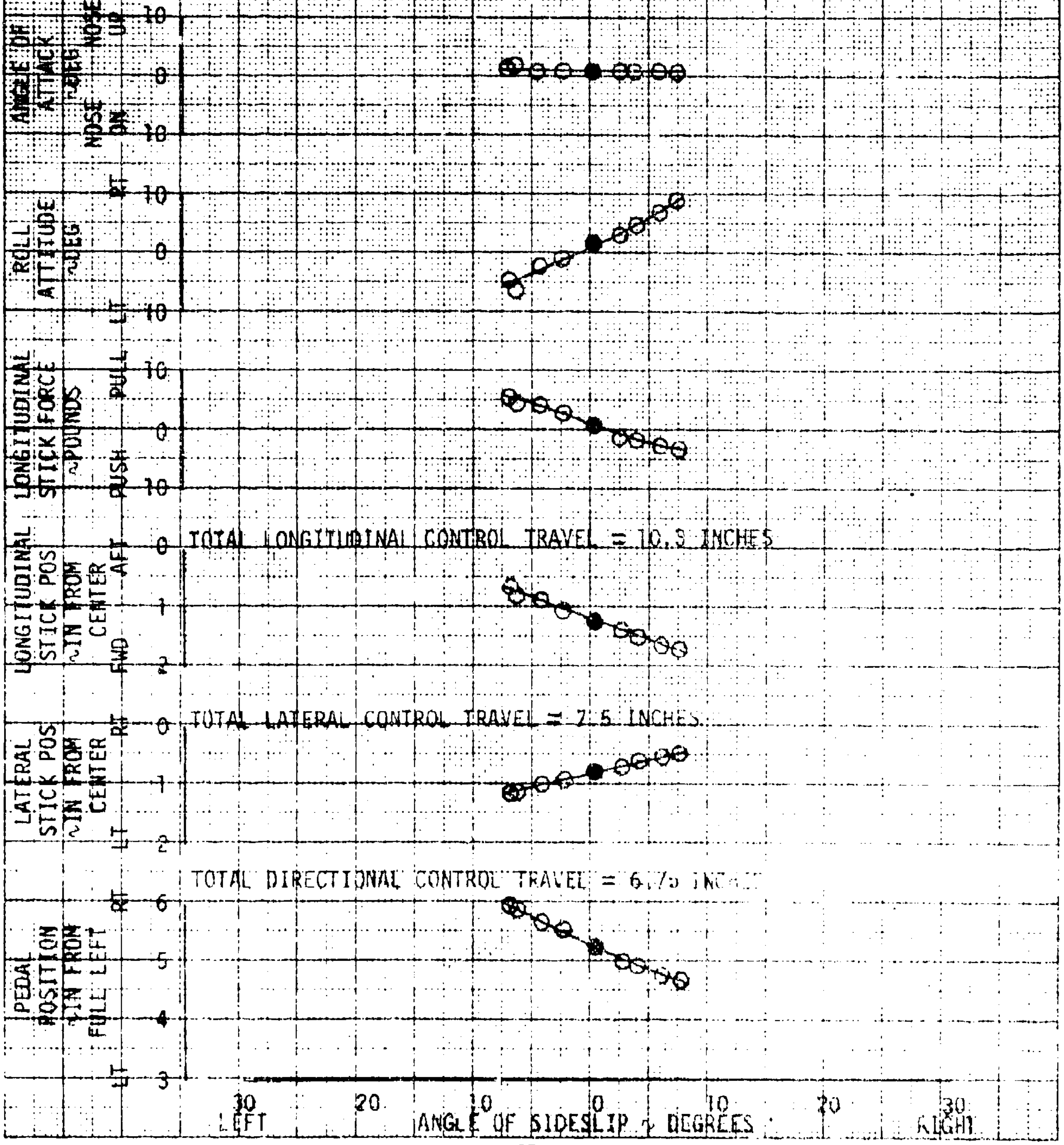


FIGURE 37
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834

AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG ROTOR SPEED ~RPM	COLL ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
19660	299.7 (AFT)	5080	249	6	150	EXT STORES (GEAR UP)

NOTE: SHADED SYMBOLS DENOTE TRIM POINTS



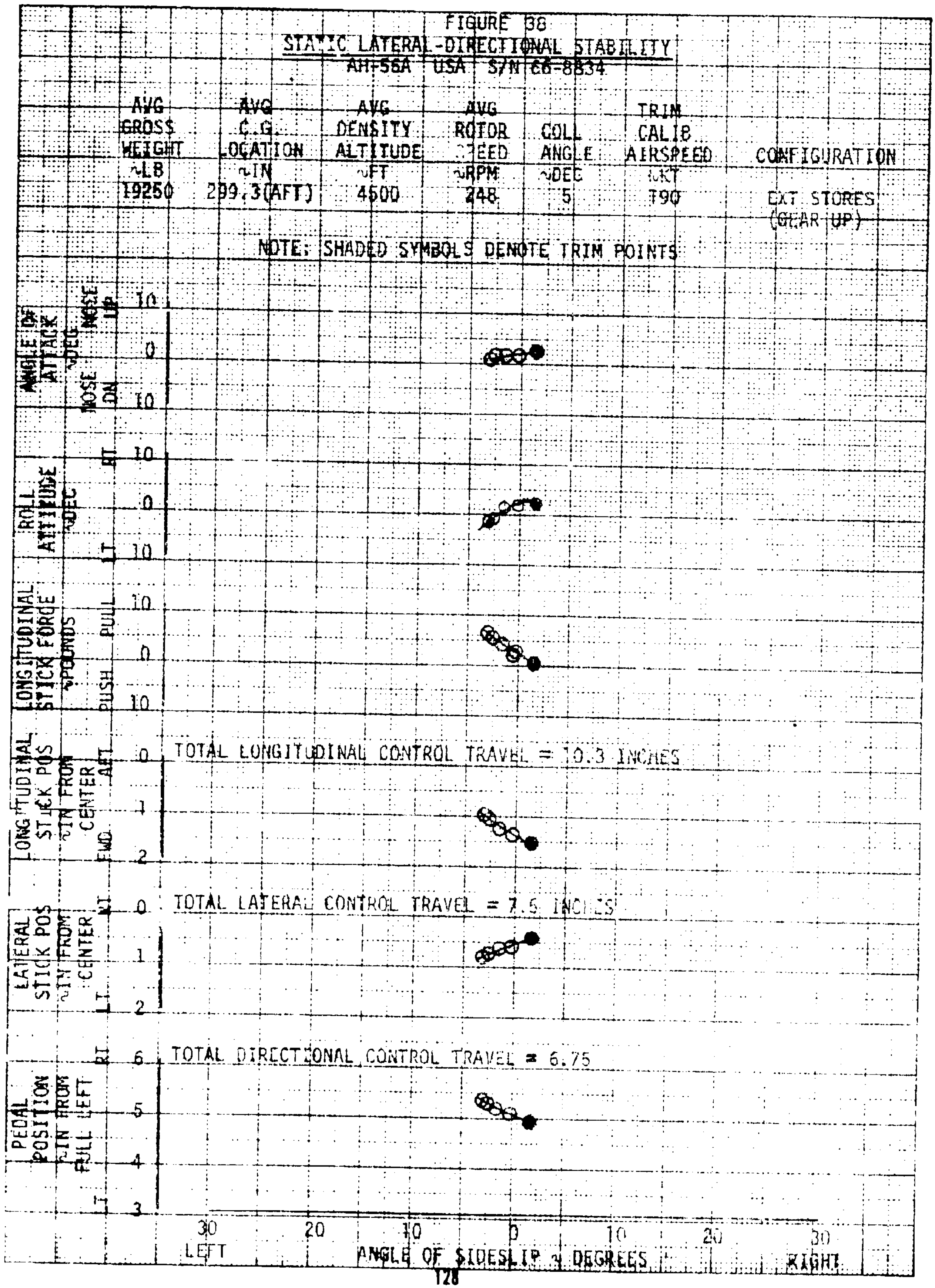


FIGURE 38
STATIC LATERAL-DIRECTIONAL STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 19250
AVG C.G. LOCATION ~IN 299.3(AFT)
AVG DENSITY ALTITUDE ~FT 4600
AVG ROTOR SPEED ~RPM 248
COLL ANGLE ~DEG 5
TRIM CALIB AIRSPEED ~KT 190
CONFIGURATION EXT STORES (GEAR UP)
NOTE: SHADED SYMBOLS DENOTE TRIM POINTS
ANGLE OF ATTACK ~DEG
NOSE UP
NOSE DN
ROLL ATTITUDE ~DEG
RT
LT
LONGITUDINAL STICK FORCE ~POUNDS
PULL
PUSH
LONGITUDINAL STICK POS ~IN FROM CENTER
AFT
FWD
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
LATERAL STICK POS ~IN FROM CENTER
RT
LT
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
PEDAL POSITION ~IN FROM FULL LEFT
RT
LT
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.75
30 LEFT
20
10
0
10
20
30 RIGHT
ANGLE OF SIDESLIP ~ DEGREES

FIGURE 39
FORWARD LONGITUDINAL PULSE
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
19400	299.8(AFT)	5940	246	122	7.1	25.8	EXT STORES (GEAR UP)

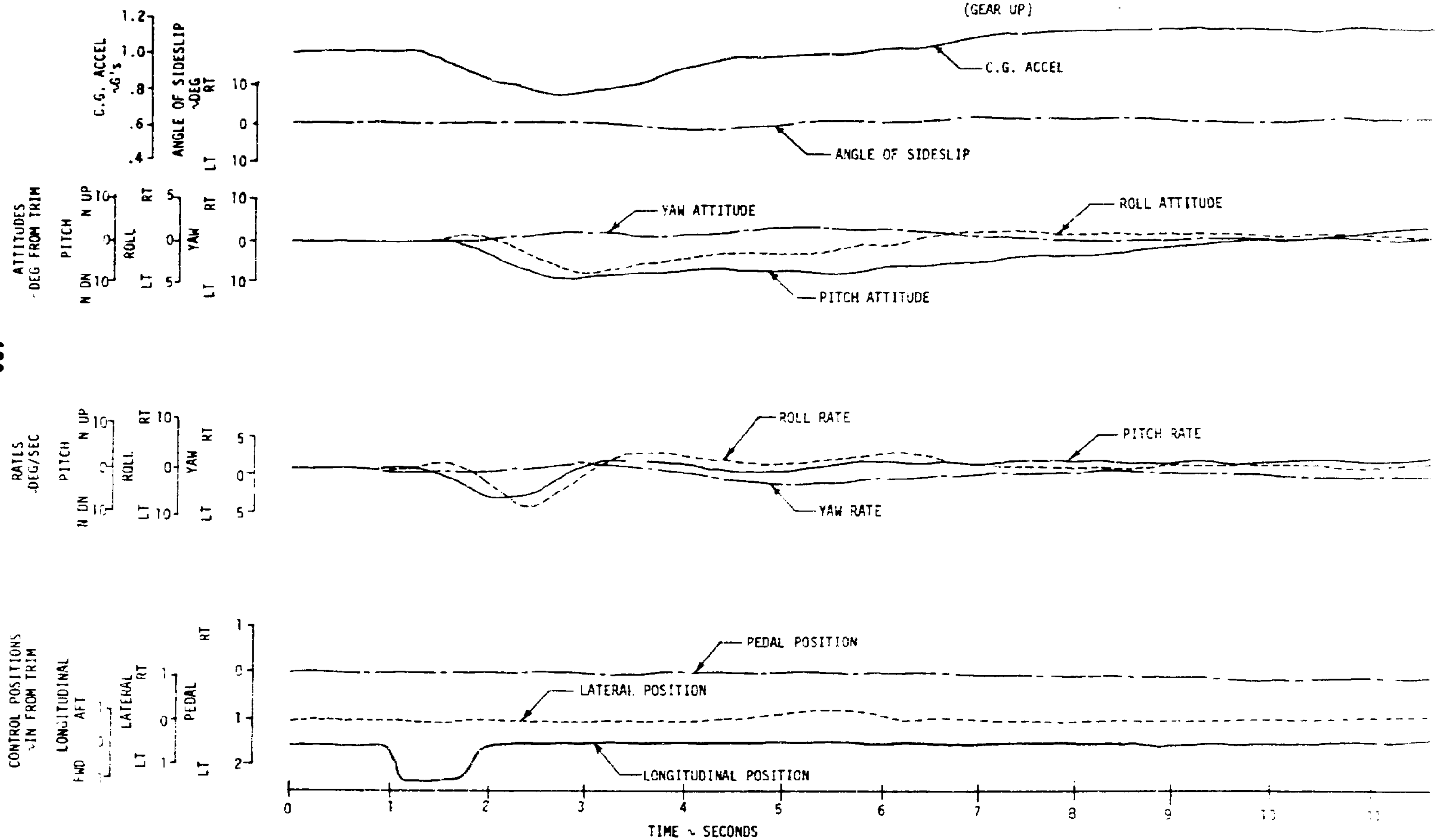


FIGURE 40
RIGHT LATERAL PULSE
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ATTITUDE ~FT	ROTOR SPEED ~RPM	CALIB AIRSPEED ~KT	COLL ANGLE ~DEG	PROP ANGLE ~DEG	CONFIGURATION
20140	239.3(AFT)	5360	246	121	5.9	25.4	EXT STORES (GEAR UP)

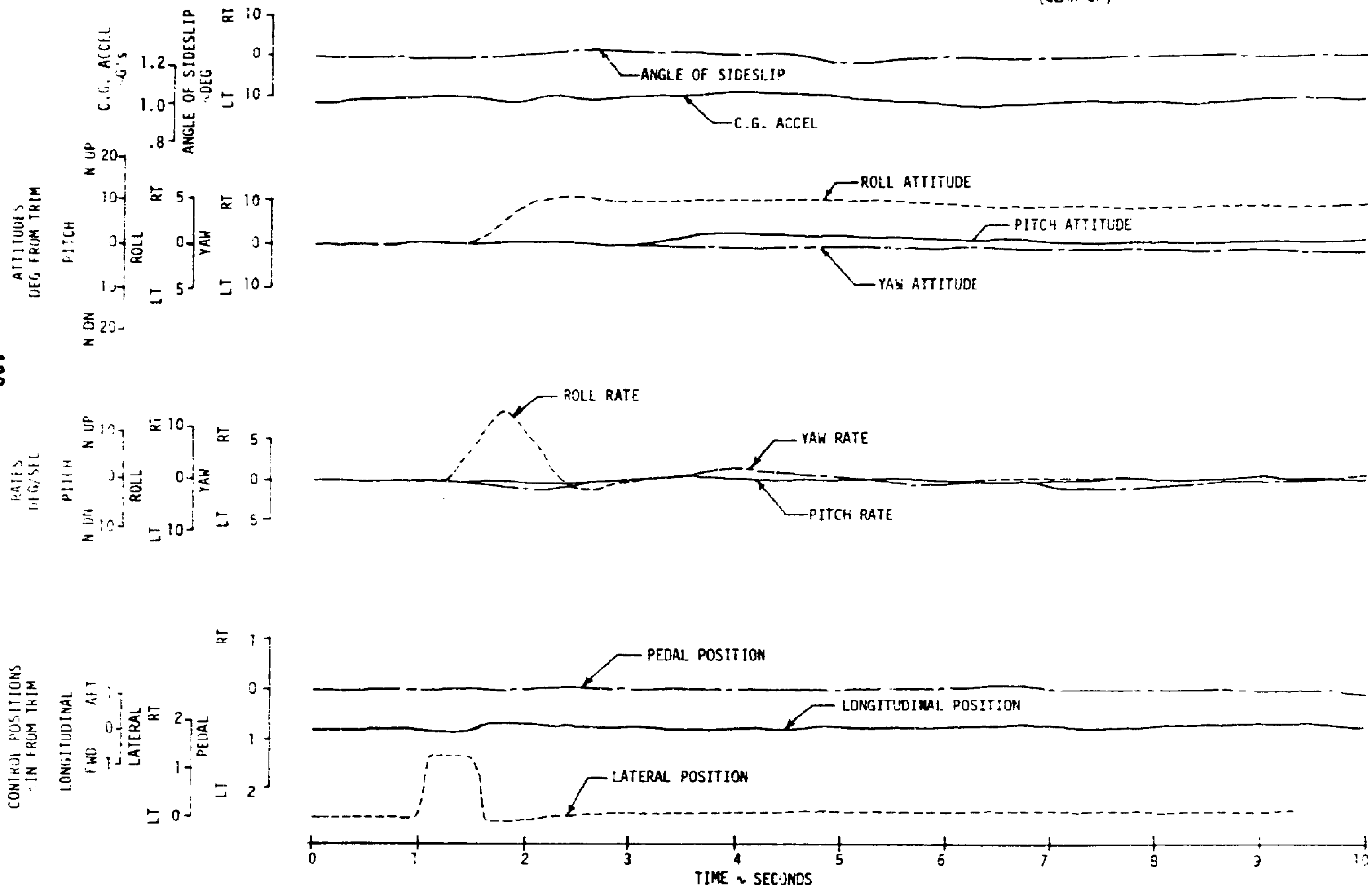


FIGURE 41
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
○	19660	299.7(AFT)	4530	248	5.1	152	EXT STORES
□	19540	299.8(AFT)	4830	248	7.0	121	EXT STORES

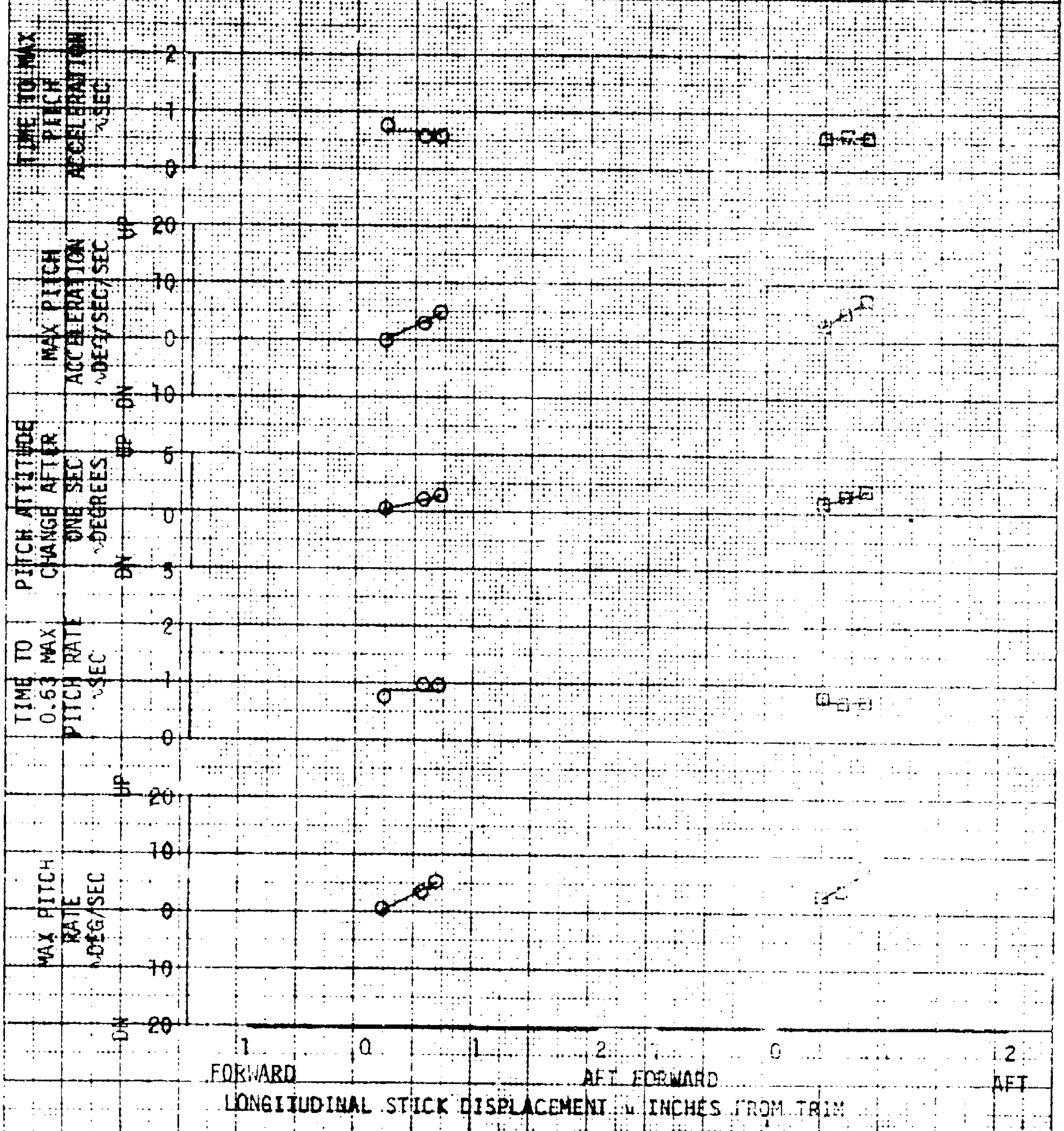
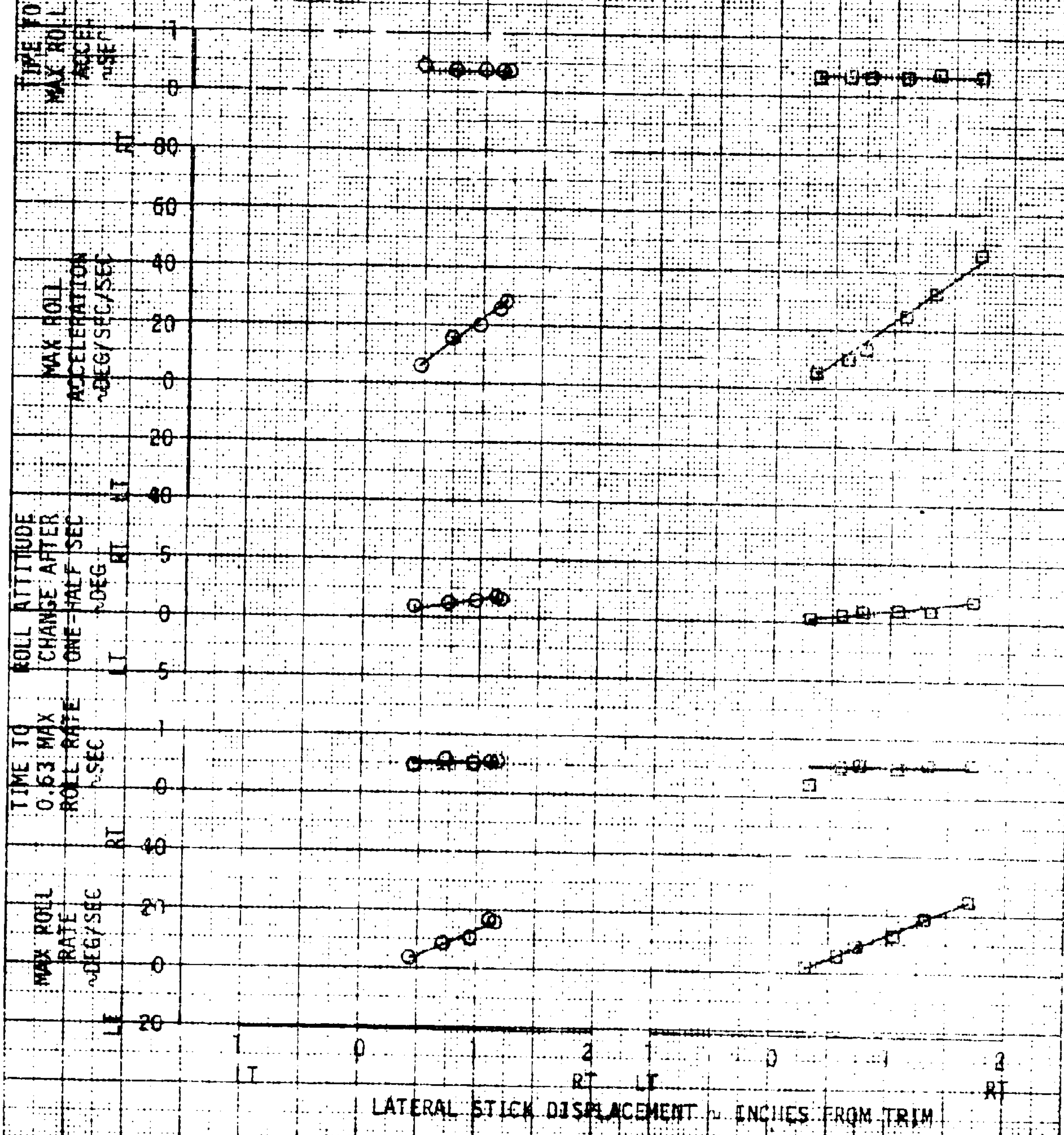


FIGURE 42
LATERAL CONTROL RESPONSE AND SENSITIVITY
AH-56A USA S/N 66-8834

SYMBOL	AVG GROSS WEIGHT ~LB	AVG C.G. LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL ANGLE ~DEG	TRIM CALIB AIRSPEED ~KT	CONFIGURATION
○	20000	299.4 (AFT)	4500	248	[illegible]	152	EXT STORES
□	20000	299.4 (AFT)	4170	247	7.0	122	EXT STORES



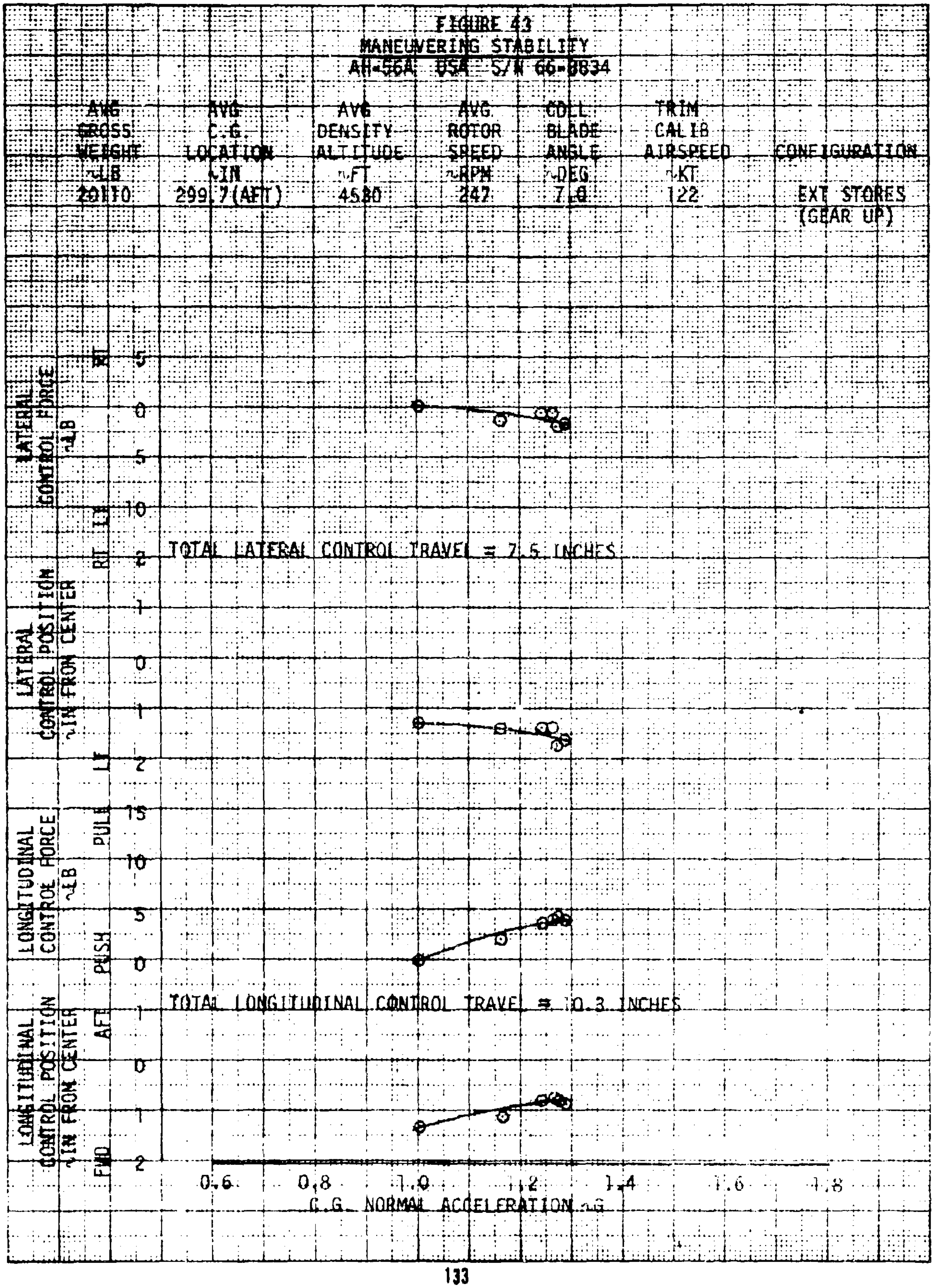


FIGURE 43
MANEUVERING STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB
20110
AVG C.G. LOCATION ~IN
299.7(AFT)
AVG DENSITY ALTITUDE ~FT
4580
AVG ROTOR SPEED ~RPM
247
COLL BLADE ANGLE ~DEG
7.0
TRIM CALIB AIRSPEED ~KT
122
CONFIGURATION
EXT STORES (GEAR UP)
LATERAL CONTROL FORCE ~LB
RT
LT
5
0
5
10
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LATERAL CONTROL POSITION ~IN FROM CENTER
RT
LT
2
1
0
1
2
LONGITUDINAL CONTROL FORCE ~LB
PULL
PUSH
15
10
5
0
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
LONGITUDINAL CONTROL POSITION ~IN FROM CENTER
AFT
FWD
1
0
1
2
0.6
0.8
1.0
1.2
1.4
1.6
1.8
C.G. NORMAL ACCELERATION ~G

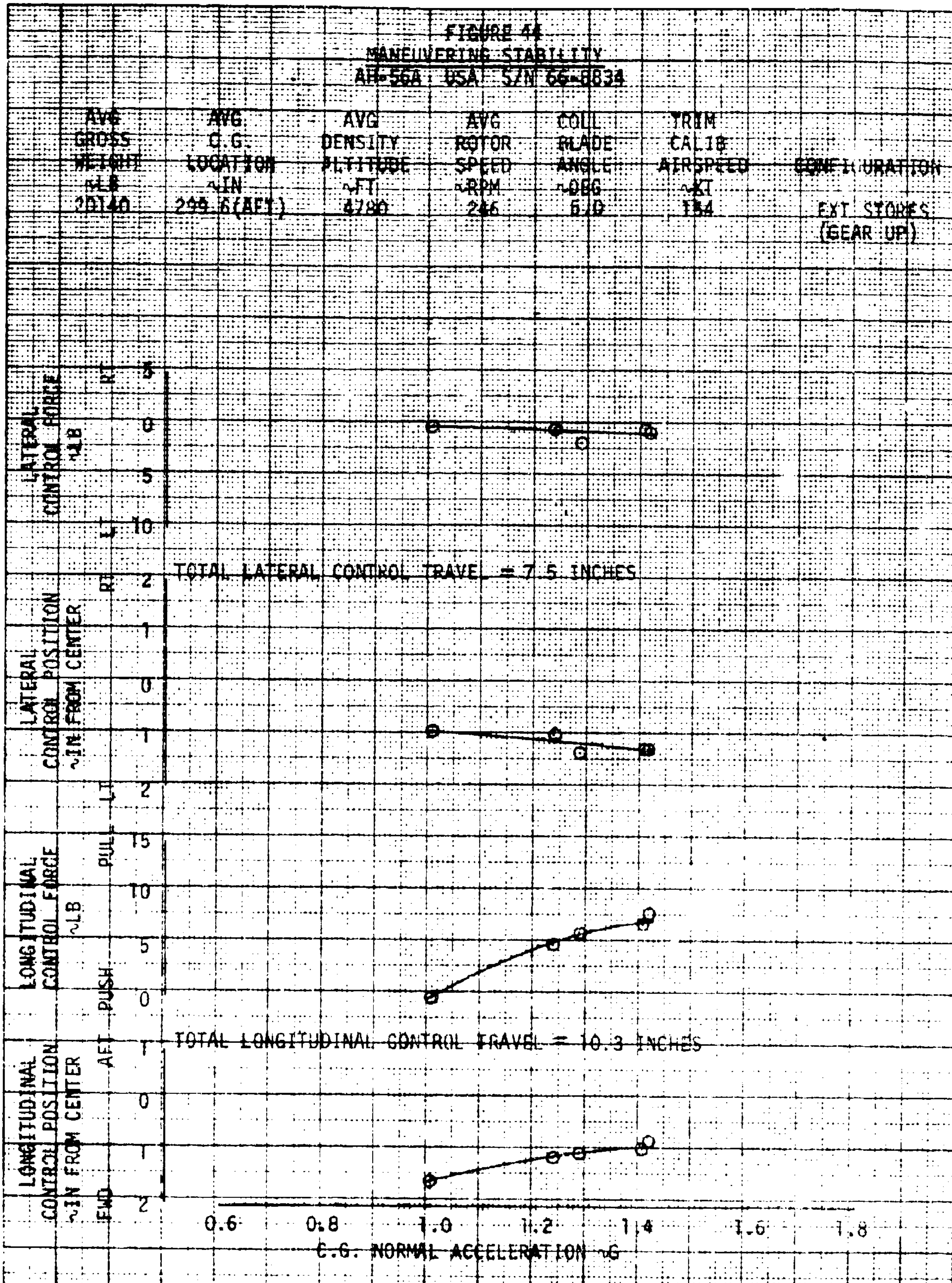


FIGURE 44
MANEUVERING STABILITY
AH-56A USA S/N 66-8834
AVG GROSS WEIGHT ~LB 20140
AVG C.G. LOCATION ~IN 299.6(AFT)
AVG DENSITY ALTITUDE ~FT 4780
AVG ROTOR SPEED ~RPM 246
COLL BLADE ANGLE ~DEG 6.0
TRIM CALIB AIRSPEED ~KT 134
CONFIGURATION EXT STORES (GEAR UP)
LATERAL CONTROL FORCE ~LB
RT
LT
TOTAL LATERAL CONTROL TRAVEL = 7.5 INCHES
LATERAL CONTROL POSITION ~IN FROM CENTER
LONGITUDINAL CONTROL FORCE ~LB
PULL
PUSH
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.3 INCHES
LONGITUDINAL CONTROL POSITION ~IN FROM CENTER
AFT
FWD
C.G. NORMAL ACCELERATION ~G

LONGITUDINAL, PITCH & C.G. ACCEL ————

PEDAL, YAW, LATERAL & ROLL - - - - - -

PEDAL, YAW & SIDESLIP —— - ——

FIGURE 45
MANEUVERING INSTABILITY
AH-56A USA S/N 66-8834

GROSS WEIGHT ~LB	C.G. LOCATION ~IN	DENSITY ALTITUDE ~FT	ROTOR SPEED ~RPM	COLL BLADE ANGLE ~DEG	CALIB AIRSPEED ~KT	CONFIGURATION
20050	300.0(AFT)	4050	246	7	120	EXT STORES (GEAR UP)

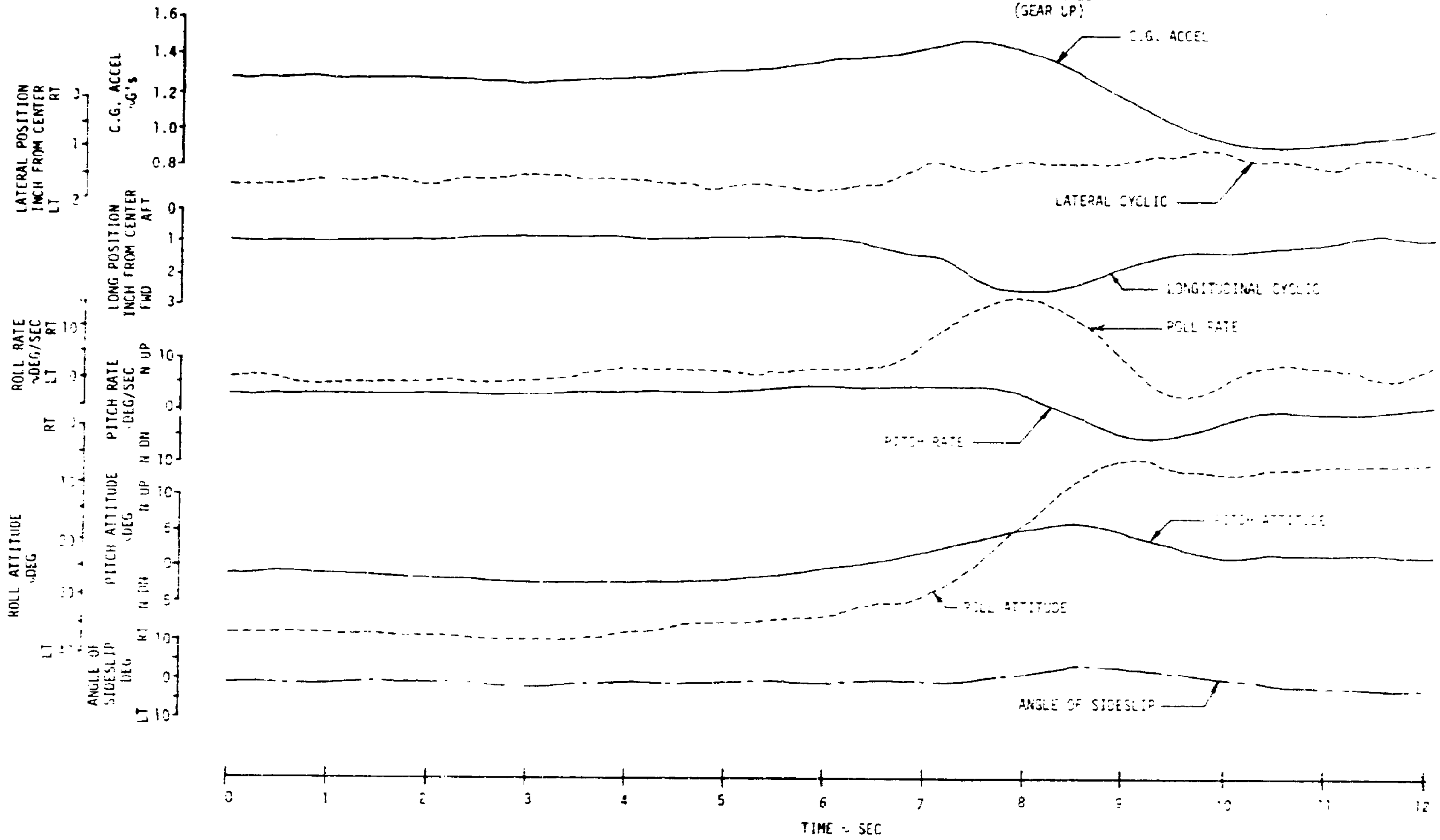
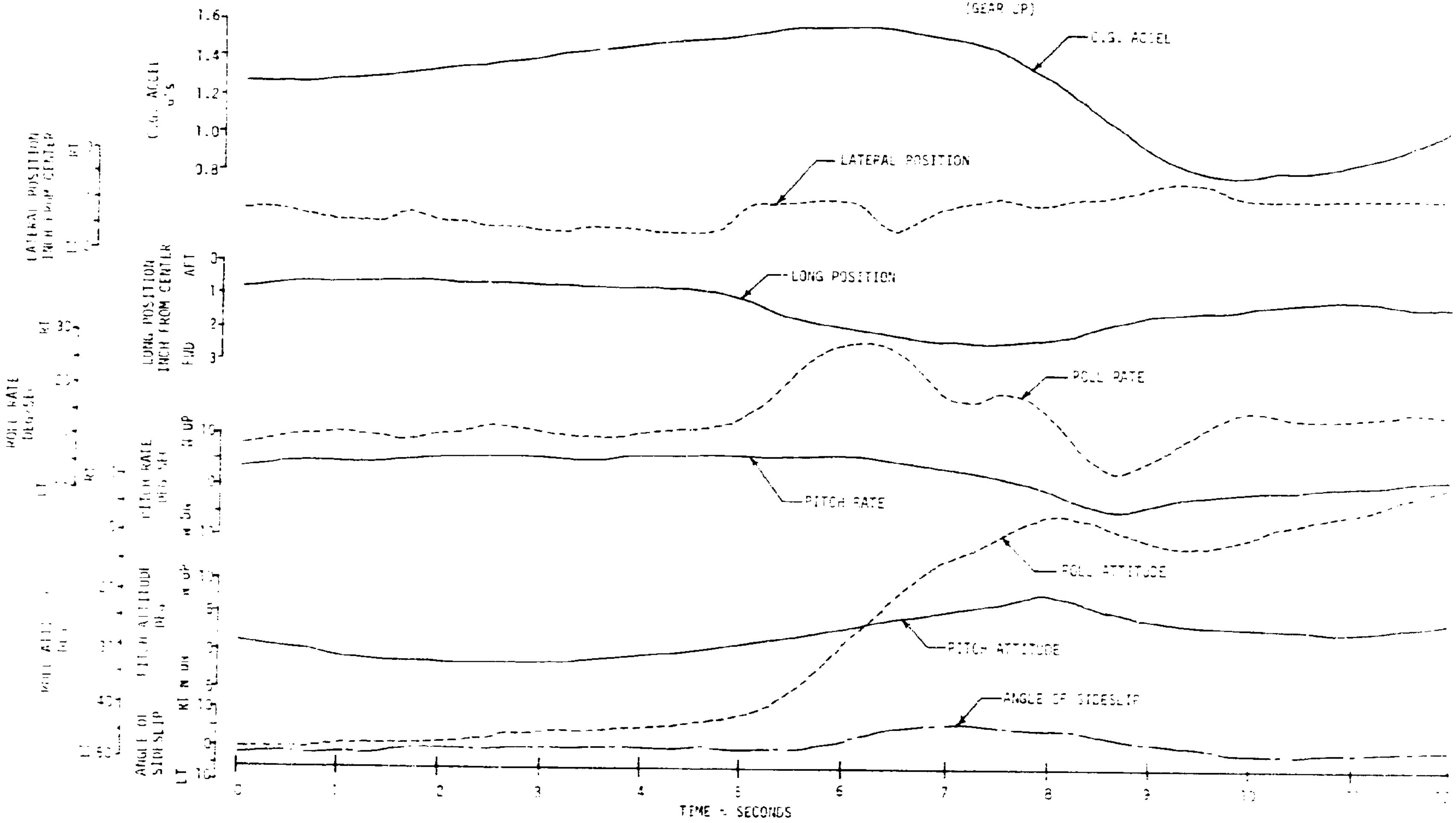


FIGURE 46
MANEUVERING INSTABILITY
AH-56A USA S/N 66-8834

LONGITUDINAL, PITCH & C.G. ACCEL ———
LATERAL & ROLL - - - - - -
PEDAL, YAW & SIDESLIP — - —

GROSS WEIGHT LB	C.G. LOCATION IN	DENSITY ALTITUDE FT	ROTOR SPEED RPM	COLL BLADE ANGLE DEG	CALIB AIRSPEED KT	CONFIGURATION
13520	299.4(AFT)	3950	246	5	145	EXT STORES (GEAR UP)



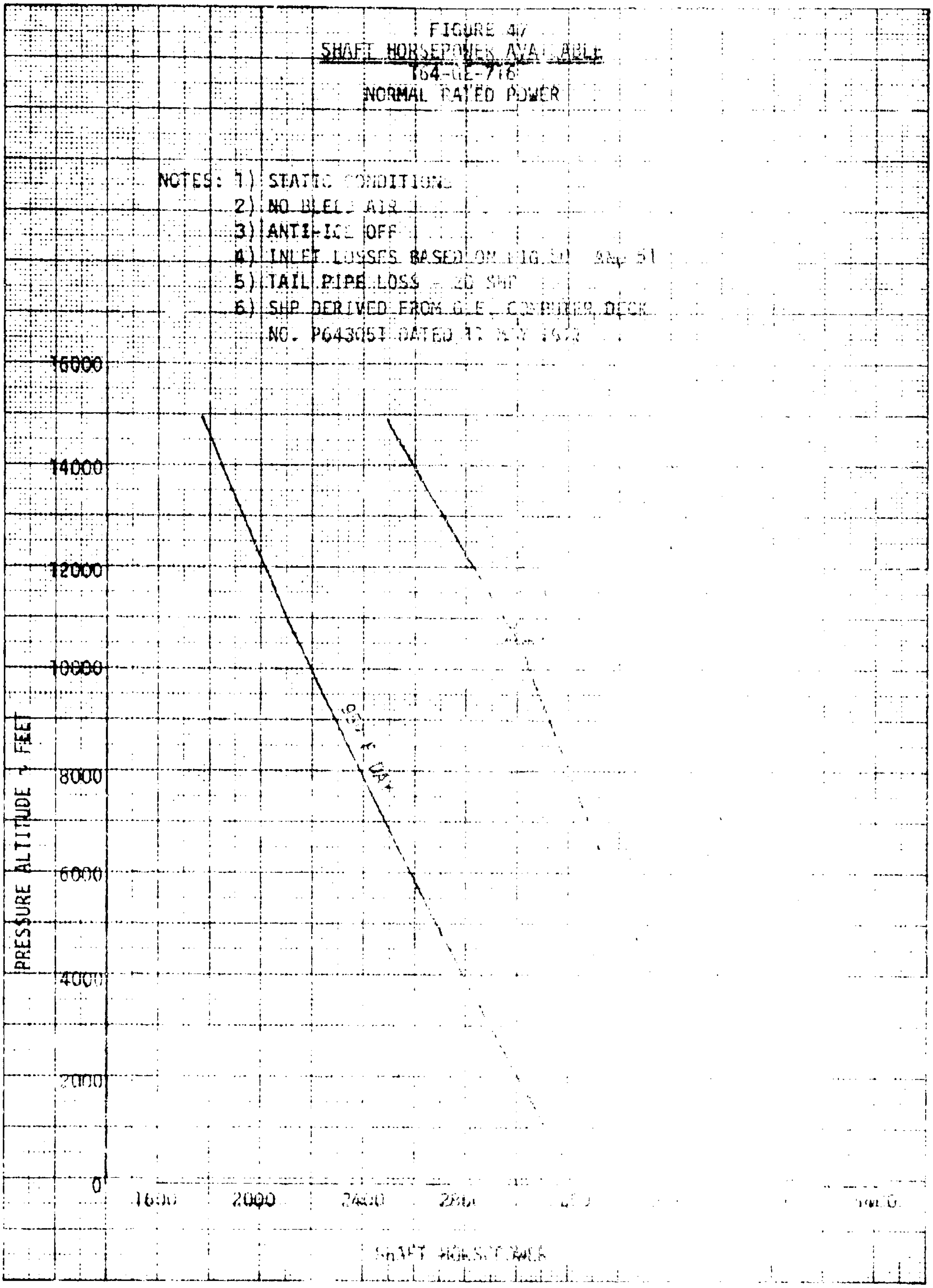


FIGURE 47
SHAFT HORSEPOWER AVAILABLE
T64-GE-716
NORMAL RATED POWER
NOTES: 1) STATIC CONDITIONS
2) NO BLEED AIR
3) ANTI-ICE OFF
4) INLET LOSSES BASED ON FIG. 50 AND 51
5) TAIL PIPE LOSS = 20 SHP
6) SHP DERIVED FROM G.E. COMPUTER DECK
NO. P643051 DATED
16000
14000
12000
10000
8000
6000
4000
2000
0
PRESSURE ALTITUDE ~ FEET
1600
2000
2400
2800
SHAFT HORSEPOWER

FIGURE 48
SHAFT HORSEPOWER AVAILABLE
T64-GE-716
MILITARY RATED POWER

NOTES: 1) STATIC CONDITIONS
2) NO BLEED AIR
3) ANTI-ICE OFF
4) INLET LOSSES BASED ON FIG 50 AND 51
5) TAIL PIPE LOSS 20 SHP
6) SHP DERIVED FROM G.E. COMPUTER DECK NO. P643051 DATED 11 MAY 1971

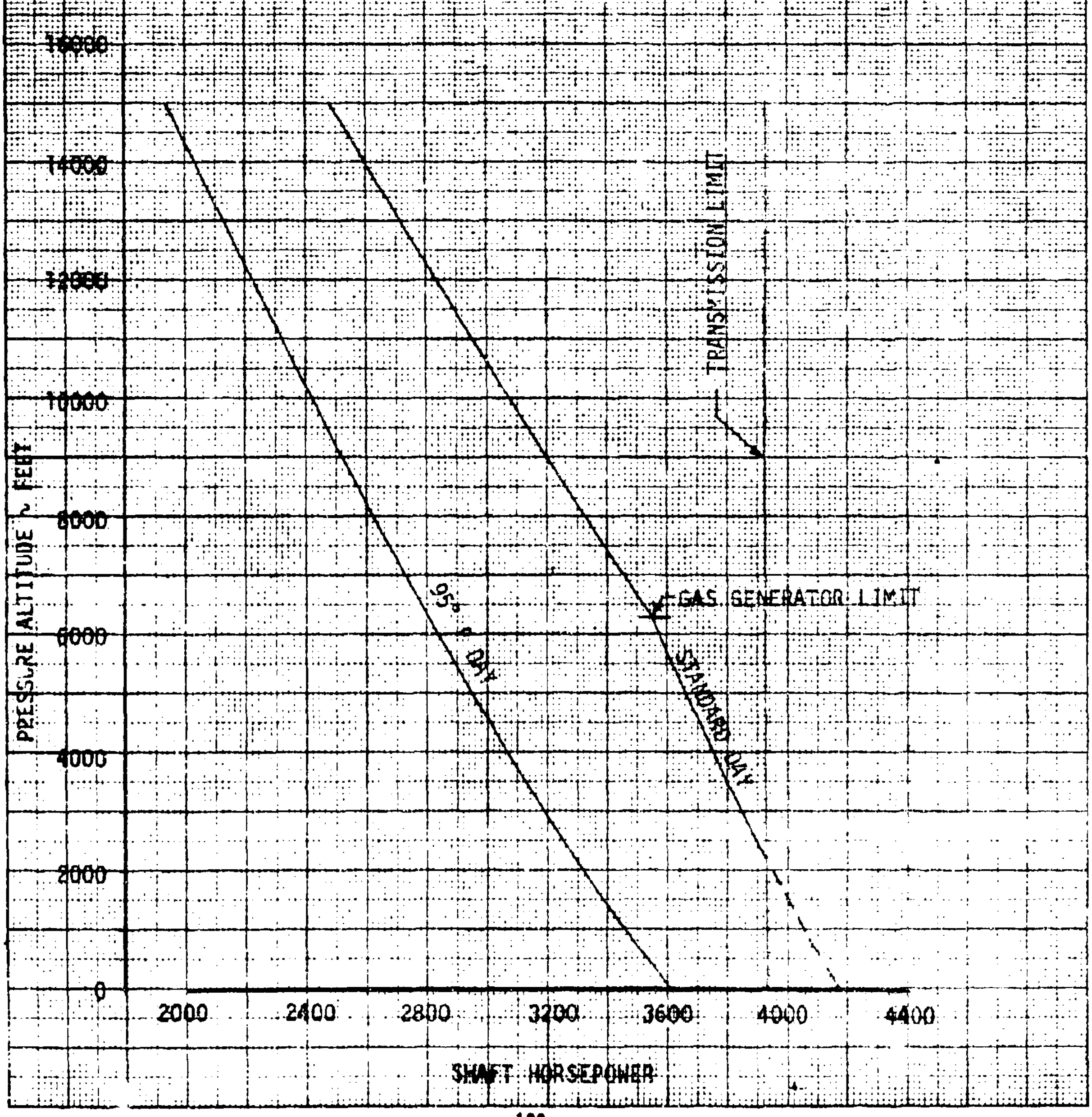
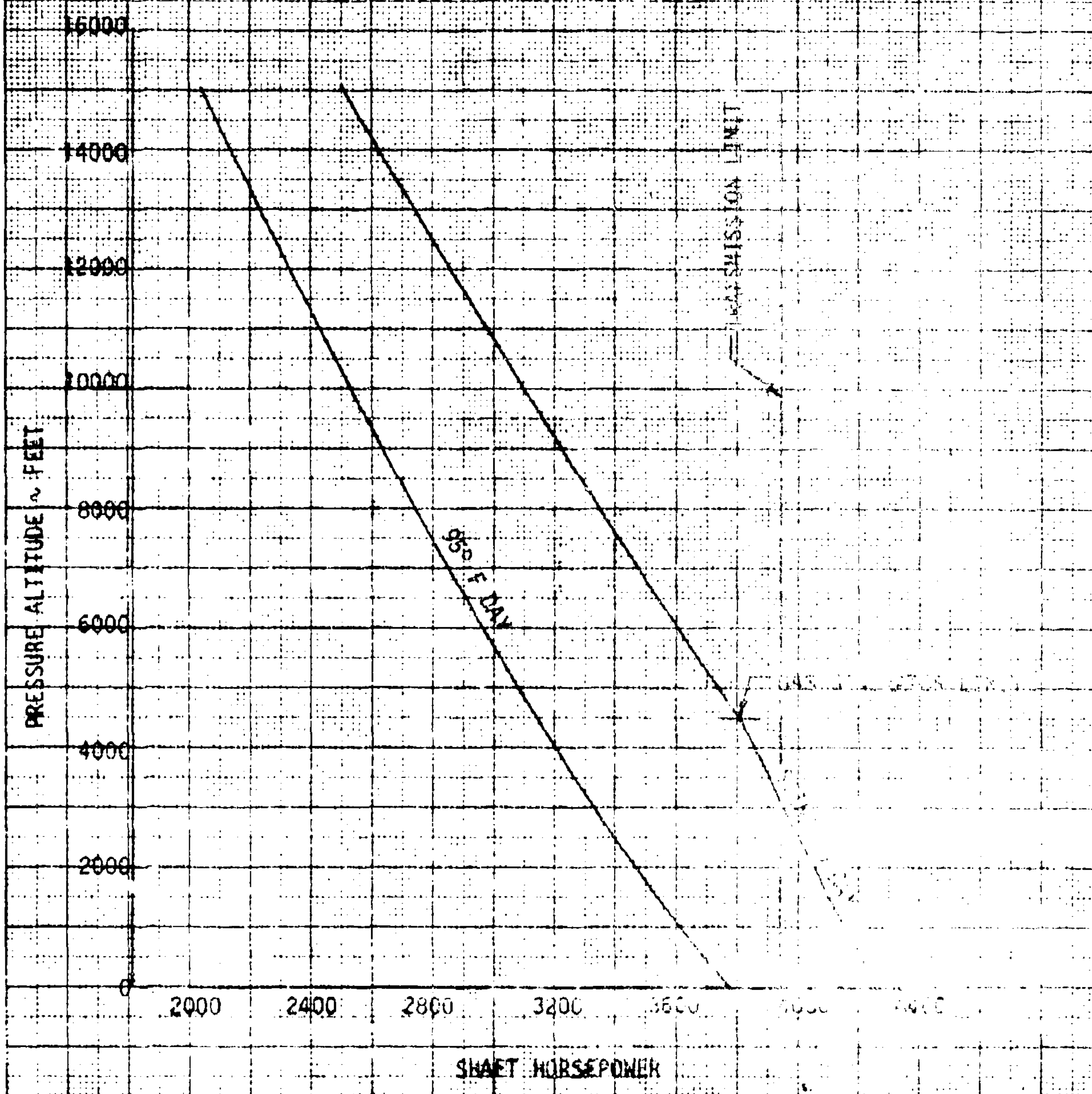


FIGURE 49
SHAFT HORSEPOWER AVAILABLE
T64-GE-716
MAXIMUM RATED POWER

NOTES:
1) STATIC CONDITIONS
2) NO BLEED AIR
3) ANTI-ICE OFF
4) INLET LOSSES BASED ON FIG 50 AND 51
5) TAIL PIPE LOSS = 20 SHP
6) SHP DERIVED FROM G.E. COMPUTER DECK NO. P643001 DATED 11 MAY 1971



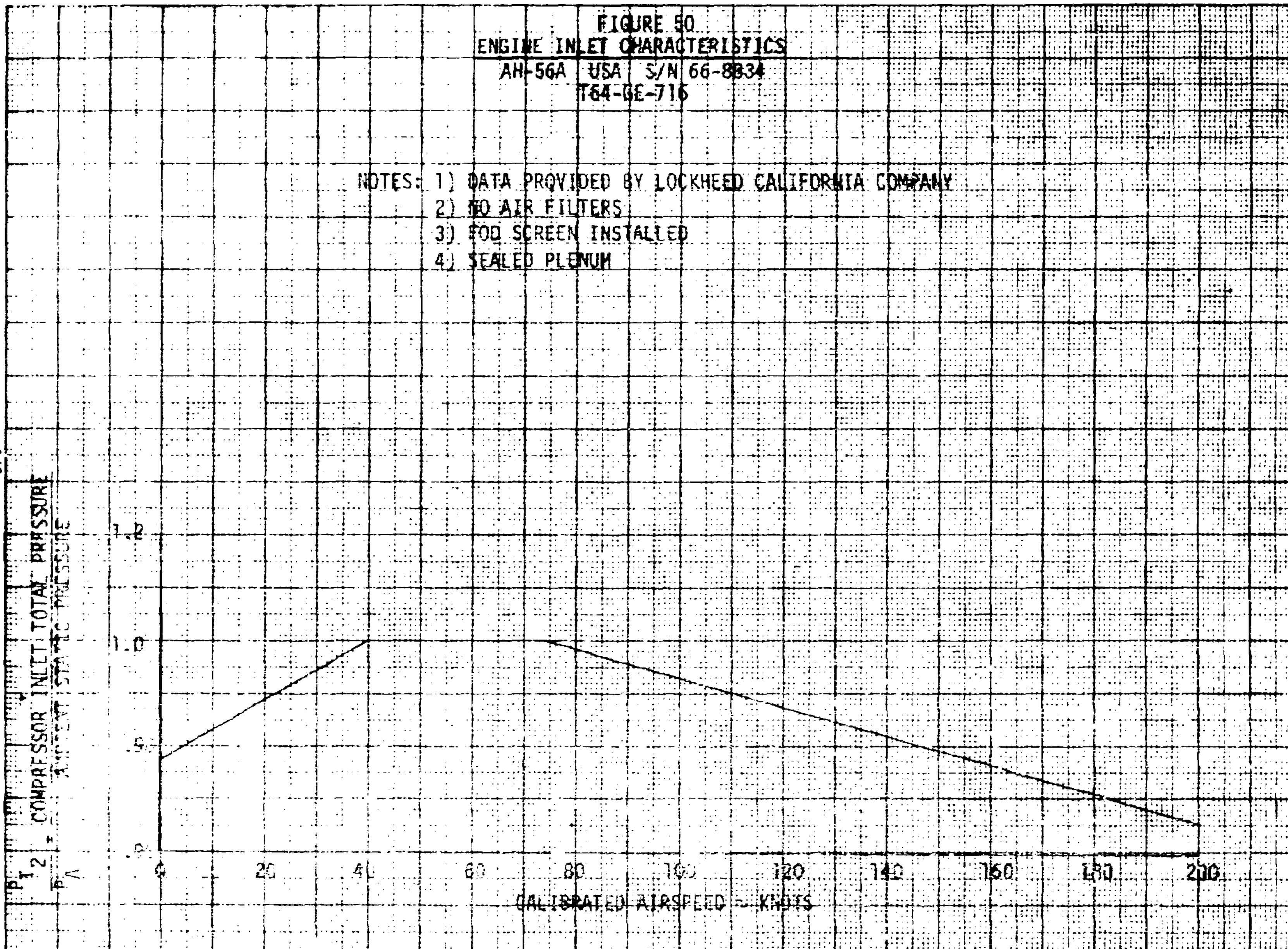


FIGURE 50
ENGINE INLET CHARACTERISTICS
AH-56A USA S/N 66-8834
T64-GE-716
NOTES: 1) DATA PROVIDED BY LOCKHEED CALIFORNIA COMPANY
2) NO AIR FILTERS
3) FOD SCREEN INSTALLED
4) SEALED PLENUM
$\frac{P_{T_2}}{P_A}$ = COMPRESSOR INLET TOTAL PRESSURE / AMBIENT STATIC PRESSURE
1.2
1.0
.9
.8
0
20
40
60
80
100
120
140
160
180
200
CALIBRATED AIRSPEED ~ KNOTS

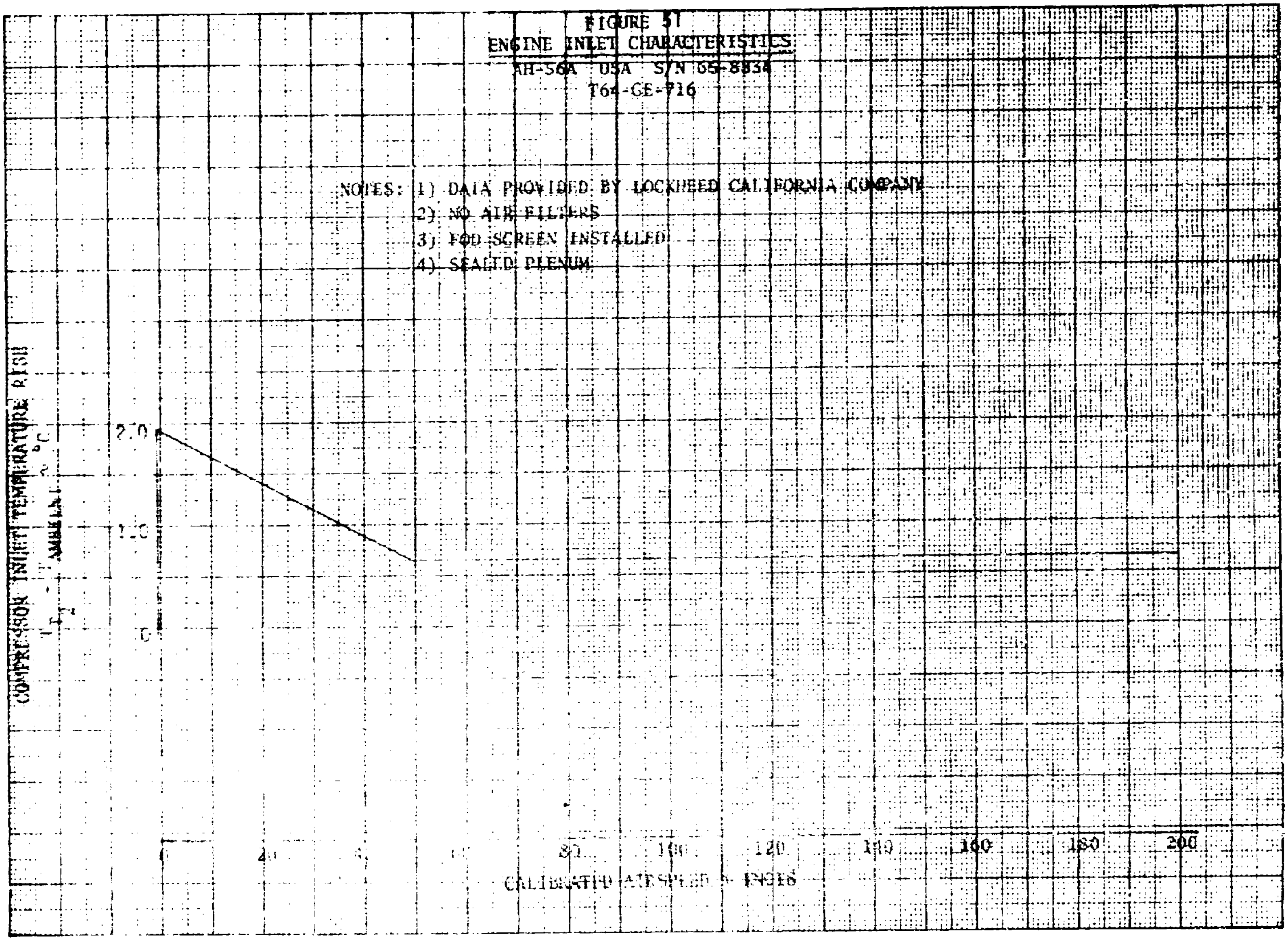


FIGURE 51
ENGINE INLET CHARACTERISTICS

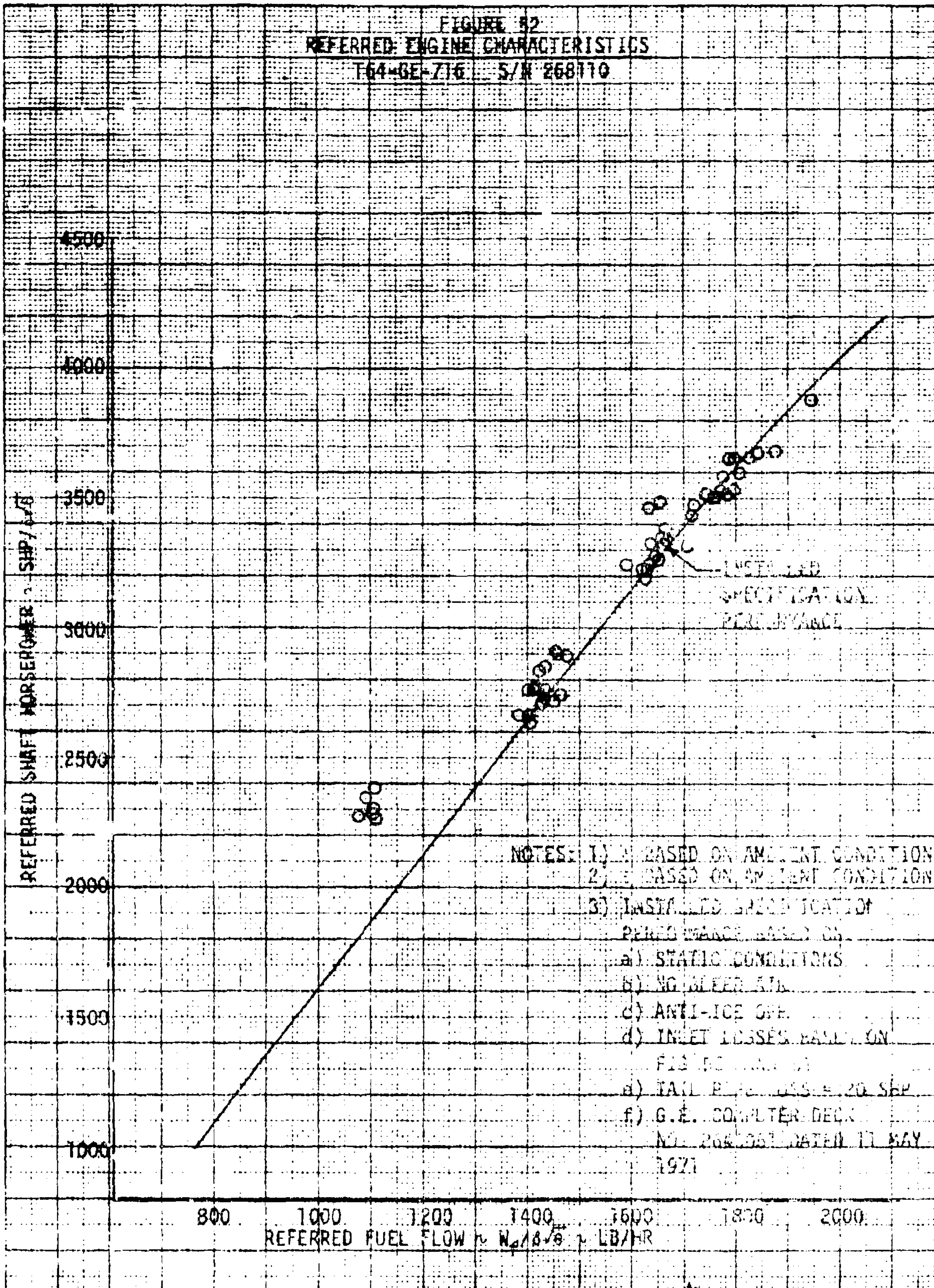


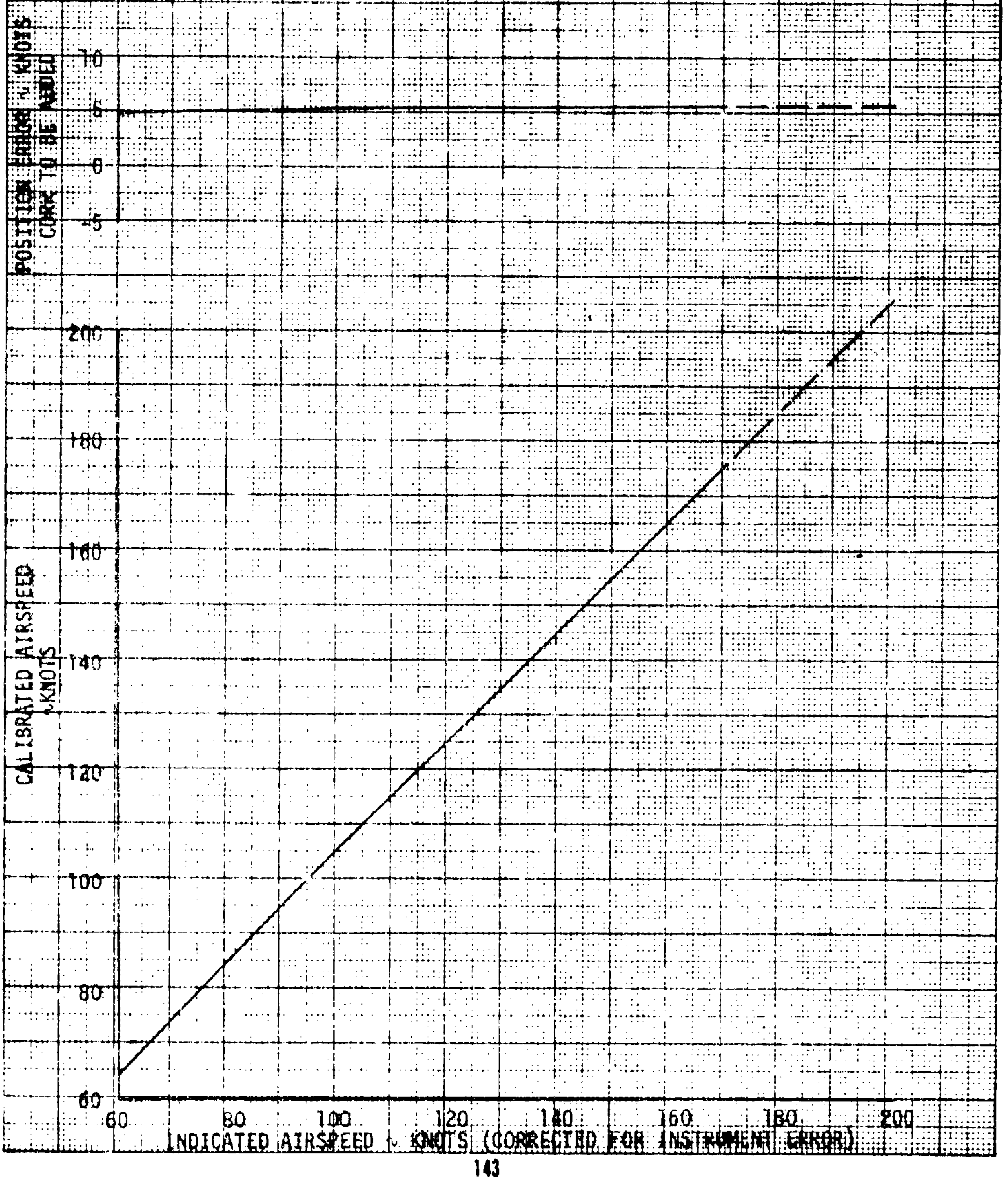
FIGURE 52
REFERRED ENGINE CHARACTERISTICS
T64-GE-716 S/N 268110
REFERRED SHAFT HORSEPOWER ~ SHP/δ√θ
4500
4000
3500
3000
2500
2000
1500
1000
800
1000
1200
1400
1600
1800
2000
REFERRED FUEL FLOW ~ $W_f/\delta\sqrt{\theta}$ ~ LB/HR
NOTES:
a) STATIC CONDITIONS
c) ANTI-ICE OFF
f) G.E. COMPUTER DECK

FIGURE 53

AIRSPEED CALIBRATION (BOOM SYSTEM)

AH-56A USA S/N 66-8834

NOTE: CALIBRATION PROVIDED BY LOCKHEED CALIFORNIA COMPANY



APPENDIX J. HUNTER-LIGGETT CDCEC 43.6/IV MISSION LOAD (12) TOW

WEIGHT EMPTY (PROPOSED PRODUCTION)	12874 ± 140
PILOT	200
CP/G	200
UNUSABLE FUEL	31
ENGINE OIL	32
XM-52 SYSTEM	757
TOW CE	80
NVS	95
ARMOR PLATE	443
OPERATING WT EMPTY	14712
PYLONS 2 @ BL 70	182
PYLONS 2 @ BL 117	176
TOW (PODS) (4)	472
CONTROL DIRECTORS (2)	12
ARM CONTROL UNITS	8
TOW MISSILES (12)	488
TOW CASES (12) (REMAIN IN PODS AFTER LAUNCH)	132
XM 52 30MM AMMO 600 RDS @ .94 #/RD	568
ZERO FUEL WT	16750
FUEL 2000#	18750