

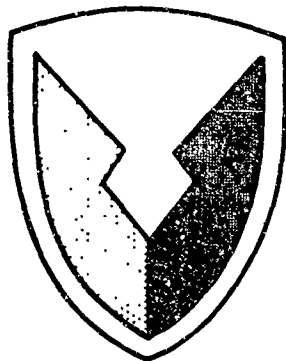
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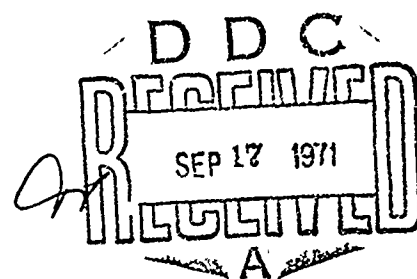
**ENGINEERING FLIGHT TEST
AH-1G (HUEYCOBRA) HELICOPTER
AUTOROTATIONAL ENTRY CHARACTERISTICS**

FINAL REPORT

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APRIL 1971



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	AH-1G HueyCobra helicopter Quantitatively evaluate response in two configurations High-air-speed/high-torque conditions 100 KCAS SCAS ON and OFF Determine differences in response Rapid roll attitude changes Rapid main rotor rpm decay rates Severity of response Fails to comply It is recommended Safe recovery is doubtful at airspeeds greater than						

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

TABLE OF CONTENTS

	<u>Page</u>
INTRODUCTION	
Background	1
Test Objectives	1
Description	1
Scope of Test	2
Method of Test	2
Chronology	3
RESULTS AND DISCUSSION	
General	4
Aircraft Response to Sudden Engine Failure	4
Roll	4
Pitch	5
Yaw and Sideslip	6
Rotor RPM Decay	6
Control Delay Time	6
Recovery Cues	7
Recovery Technique	7
Military Specification Compliance	8
Response Limits	9
SCAS OFF	10
CONCLUSIONS	11
RECOMMENDATIONS	13
APPENDIXES	
I. References	14
II. Basic Aircraft Information and Operating Limits	15
III. Test Instrumentation	23
IV. Test Data	26
V. Distribution	54

ABSTRACT

Engineering flight tests were conducted to quantitatively evaluate the response of the AH-1G HueyCobra helicopter to simulated sudden engine failures at high airspeed and high engine power combinations in two configurations. Additionally, simulated engine failures were investigated at 100 knots calibrated airspeed (KCAS) with the stability and control augmentation system (SCAS) ON and OFF to determine the differences in aircraft response with an inoperative SCAS. Aircraft response to simulated sudden engine failure was characterized by rapid roll attitude change and rapid main rotor rpm decay rates, both of which were unacceptable at the maximum engine torque settings. The severity of the aircraft response is primarily a function of engine torque at the time of failure. The AH-1G helicopter fails to comply with both the present and proposed military specifications for flying qualities of helicopters with regard to autorotational entry at high-airspeed/high-torque conditions. The present military specification, MIL-H-8501A, does not fully or realistically prescribe a safe operating limit for this class of helicopter. It is recommended that during daytime, visual flight conditions the AH-1G not be operated at combinations of engine torque and airspeed where the pilot recognition and reaction time is less than 1.5 seconds, and the bank attitude change is more than 40 degrees in 2 seconds following a sudden engine failure. It is also recommended that during night and limited visibility conditions the AH-1G be further limited to those conditions which safely allow a 2-second control delay. Aircraft response to sudden engine failure with the SCAS inoperative is more severe, and safe recovery is doubtful at airspeeds greater than 100 KCAS with high engine torque settings.

INTRODUCTION

BACKGROUND

1. The autorotational entry characteristics of the AH-1G helicopter following simulated sudden engine failure have been reported both as marginally acceptable and unacceptable for the high airspeed, maximum engine power conditions (refs 1 through 7, app 1). During testing of the AH-1G with the stabilized night sight (SNS) installed, it was qualitatively determined that the severity of the aircraft motions following simulated engine failures at high airspeeds was significantly reduced at indicated engine torque values less than 35 pounds per square inch (psi). Consequently, it was recommended that the engine torque be limited to 35 psi for all airspeeds greater than 150 knots calibrated airspeed (KCAS) for all AH-1G configurations tested (refs 3 through 7). The US Army Aviation Systems Test Activity (USAASTA) was directed by the US Army Aviation Systems Command (AVSCOM) to conduct flight tests on the AH-1G helicopter to determine the effects of reduced engine torque settings on the autorotational entry characteristics following simulated sudden engine failures at high airspeeds (ref 8).

TEST OBJECTIVES

2. The objectives of this test were to determine the response of the AH-1G helicopter following simulated sudden engine failure at airspeeds greater than 140 KCAS for various entry engine torque values and to determine if the use of reduced engine torque settings would enhance the safety of high airspeed operations.

DESCRIPTION

3. The AH-1G helicopter, manufactured by Bell Helicopter Company (BHC), was developed to meet the US Army requirements for an armed helicopter. Tandem seating is provided for the two-man crew. The main rotor system is a two-bladed, semirigid, "door hinge" type. An antitorque rotor is located near the top of the vertical stabilizer. A three-axis stability and control augmentation system (SCAS) is provided to improve the aircraft handling qualities. The helicopter is powered by a Lycoming T53-L-13 turboshaft engine rated at 1400 shaft horsepower (shp) at sea-level (SL), standard-day, uninstalled conditions, but usable power is 1100 shp because of a main transmission maximum torque limit. Distinctive features of the AH-1G are the narrow fuselage (36 in.), the stub midwing with four external store stations, and the integral chin turret. The flight control system is of the mechanical, hydraulically boosted, irreversible type with conventional helicopter controls in the aft cockpit (pilot station). The controls in the forward cockpit (copilot/gunner station) consist of conventional antitorque pedals and sidearm collective and cyclic controls. An electrically operated force trim system is connected to the cyclic and

directional controls to provide artificial feel and positive control centering. The elevator is synchronized with the cyclic stick. The pilot normally fires the wing stores and can fire the chin turret in the stowed position. The copilot/gunner operates the flexible turret and can also fire the wing stores. The wing stores can be jettisoned by either the pilot or gunner in case of emergency. The design gross weight (grwt) for the AH-1G is 6600 pounds, and the maximum gross weight is 9500 pounds. The test helicopter, S/N 66-15247, was representative of a standard production AH-1G with certain exceptions not affecting the flight characteristics. Both instrument panels were extensively modified to accommodate the test instrumentation. This aircraft was the same aircraft used for the autorotational entry tests (refs 1 and 3, app I). A detailed description of the test aircraft is contained in appendix II. The test instrumentation used during these tests is listed in appendix III.

SCOPE OF TEST

4. The test program consisted of a quantitative evaluation of the autorotational entry characteristics at two gross weights, two center-of-gravity (cg) locations, and two wing stores configurations. These configurations were determined during the previous tests to result in the most adverse aircraft reactions. These configurations consisted of a 9200-pound mean grwt at a 192.7-inch forward cg location with four XM159 rocket pods installed, and a 7350-pound mean grwt at a 201.0-inch aft cg location with clean wings. The tests were conducted at 140, 150, 160 and 170 knots calibrated airspeed (KCAS) for five entry indicated engine torque values between 25 and 49 psi at each airspeed. Additionally, the autorotational entry characteristics were evaluated in 30-degree left bank turns at 140 KCAS and in wings-level flight at 100 KCAS with the SCAS both ON and OFF. These additional tests were conducted in the light weight, aft cg, clean wing configuration only. All entries were initiated at an approximate 5000-foot density altitude (H_D). The test program required four flights and 5 hours to complete.

METHOD OF TEST

5. The method of test used was to establish stabilized flight at the desired entry airspeed and engine torque values. Sudden engine failure was simulated by rapidly closing the twist grip throttle. All flight controls were held fixed in the power-on trim positions until the maximum tolerable aircraft attitude, rate, acceleration or rotor speed decay was observed. For each test point, data were continuously recorded on the oscillograph and voice tape from trimmed power-on flight until the aircraft was stabilized in normal autorotational flight.

CHRONOLOGY

6. The chronology of the test program is as follows:

Test directive received	4	August	1970
Test started	1	December	1970
Test completed	9	December	1970

RESULTS AND DISCUSSION

GENERAL

7. The results of these tests indicate that the response of the AH-1G helicopter, following sudden engine failure at high airspeed and high engine torque conditions, is severe and unacceptable for safe flight operations. The limit of tolerable aircraft response was reached in a very short time interval for the airspeed and engine torque conditions tested. The short delay times do not provide sufficient recognition and reaction time following an actual sudden engine failure for operational pilots. All simulated engine failures were planned and controlled by the evaluation pilot, who was experienced in the sudden engine failure maneuver and familiar with the aircraft response and recovery characteristics for each test condition. The results indicated that the severity of the AH-1G response following sudden engine failure was strongly influenced by engine torque at the time of failure. Airspeed had a somewhat less, but significant, influence on the aircraft response. The aircraft response following sudden engine failure in steady, 30-degree banked left turns was similar to that recorded for wings-level entry at the same airspeed/engine torque values. However, the delay time in the left turn was less because the limit left bank angle was reached sooner. The tests with SCAS OFF at 100 KCAS show a large increase in the severity of the aircraft response as compared to that with SCAS ON at the same airspeed and engine torque values. Engine torque restrictions are required at airspeeds greater than 140 KCAS to provide a reasonable margin of safety for the crew and aircraft.

AIRCRAFT RESPONSE TO SUDDEN ENGINE FAILURE

8. The aircraft response following the simulated sudden engine failure is shown in figures 1 through 23, appendix IV. In all cases, the response is similar, varying only in severity. Measured control delay times between simulated engine failure and recovery control input are shown in figures 24 and 25. The aircraft response (attitudes, rates and accelerations) was determined at the time of recovery. Since the test was conducted without preestablished values to define the attitude, rate, or acceleration at which recovery was to be made, considerable scatter was expected in the control delay times. The scatter is considered small for this type of test. Consequently, the aircraft response information derived at those delay times is considered a valid representation of the effects of varying entry engine torque and airspeed. The time histories presented in figures 20 through 23 are typical of the various conditions and configurations evaluated.

Roll

9. The roll response of the aircraft to simulated sudden engine failure is shown in figures 1 through 8, appendix IV. The roll attitude at recovery shows a definite trend to increase, peak, and then decrease as engine torque at entry increases.

The engine torque value at which the curve peaks decreases with increasing airspeed. The maximum roll attitude, before completion of recovery to steady autorotation, was plotted for all conditions of entry engine torque and airspeed. Where the roll attitude at recovery curve had peaked or was decreasing, the maximum roll attitude was constant at 50 ± 5 degrees. For flight maneuvers with less than 1g normal acceleration, there exists a bank attitude which cannot be safely exceeded with a helicopter. The rapid rotor rpm decay also affected the test maneuver bank attitude limit. Although loss of control was not experienced during these tests, it is felt that the bank attitudes, low rotor speeds, and dive angles encountered are unacceptable for other than controlled test conditions.

10. The left roll rates, 1 second after simulated engine failure, show little change with increasing airspeed, some expected difference with configuration change, and a significant increase with increasing engine torque (fig. 4, app IV). The left roll rates at initiation of recovery show insignificant change with increasing airspeed and configuration. With increasing engine torque, the roll rate at recovery shows a definite trend to peak and then decrease while the maximum roll rate experienced during the maneuver continues to increase. Analysis of these trends suggests that the pilot was integrating the roll rate to effect recovery at the limit bank attitude of approximately 50 degrees. Where the roll rate at recovery curve peaked, the pilot may have integrated the roll acceleration to control the attitude during the maneuver. Figure 8 shows acceleration to be a function of engine torque with some airspeed and configuration effects.

11. The roll rates and accelerations recorded correspond to those resulting from a lateral input of 2 inches in powered flight. The trim curves in level flight and stabilized autorotation at 100 KCAS show a lateral trim difference of approximately 2 inches (ref 3, app I). This static trim shift is to the right when going from powered flight to autorotation. No static trim curves are available for autorotation at airspeeds greater than 100 KCAS, but lower airspeed trends support a similar trim difference. It is significant that this trim shift occurs simultaneously with the loss of positive torque to the rotor. These data indicate that right lateral control movement is required to control the left roll response. The magnitude of the input varied with the entry engine torque and airspeed; and at the maximum entry engine torque values, the lateral control movements required to check the roll rates were approximately 1 1/2 inches.

Pitch

12. The aircraft pitch attitude, pitch rate and normal acceleration are not presented, except in time histories. These parameters were analyzed from the records and were determined to have an insignificant effect on the pilot's recognition of the requirement for recovery action. The aircraft pitch rate was essentially zero, as seen in figures 20 through 23, appendix IV, until a longitudinal cyclic input was made to effect recovery. The changes in normal acceleration were small and are attributed to the reduction of the vertical thrust component (lift) as the rotor rpm decayed. Normal acceleration (g force) and pitch rate changes were not perceived by the pilot.

Yaw and Sideslip

13. The maximum yaw rate, yaw rate at recovery, yaw rate at 1 second, maximum angle of sideslip, and sideslip angle at 1 second are presented in figures 9 through 14, appendix IV. The time to maximum yaw rate was essentially constant between 0.3 and 0.4 second, regardless of entry condition. The maximum yaw rate was 8 to 12 deg/sec for all airspeeds, configurations, and entry engine torques above 35 psi. This yawing was not perceived by the pilot at the high airspeeds because the roll response dominated the pilot's attention. The sideslip resulting from the simulated engine failures at the higher engine torque values exceeded the sideslip envelope of the aircraft. During an entry from 160 KCAS and 48 psi engine torque, a Dutch-roll type lateral-directional oscillation was experienced. Approximately 4 cycles were observed before the sideslip was reduced by pedal input during recovery. This oscillation was not observed on any of the other entries.

Rotor RPM Decay

14. The rotor rpm decay characteristics are shown in figures 15 through 18, appendix IV. For the control delay times achieved during these tests, the minimum transient rotor speed allowed, 250 rpm, was approached on several occasions. Entry from 150 KCAS and 46 psi engine torque resulted in the minimum rotor speed reached during the tests, 248 rpm. Although the rotor rpm at recovery was higher at the higher entry engine torque values, the increased decay rate resulted in the same minimum rotor speeds, 250 to 260 rpm. The decay rates are presented in figure 18. The results show a linear increase in decay rate with increasing engine torque with some gross weight and cg effects and little airspeed effect. The maximum rotor rpm decay rates were 32.5 rpm/sec in the light weight, aft cg, clean wing configuration. The extrapolated decay rate for full power, 50 psi engine torque, is 35 rpm/sec in this configuration. The rotor rpm value was not the prime recovery cue reported during these tests. However, it is obvious that if the severe roll attitude changes were not present, the minimum allowed transient rotor rpm would require recovery action with little increase in the usable delay time.

CONTROL DELAY TIME

15. The maximum control delay time was determined at each test point by holding the controls fixed in the trimmed powered-flight positions until the limit of tolerable aircraft response was reached. The control delay time represents the time interval between engine failure (rotor rpm starting to decrease) and the first control input to effect recovery. The control delay times for all test conditions are shown in figures 24 and 25, appendix IV. For all high-airspeed conditions, the critical control was the cyclic. The cyclic motion required to initiate recovery was aft and right. The data plotted in figures 24 and 25 show the strong effect of entry engine torque as well as the lesser, but significant, effect of airspeed on the control delay time.

RECOVERY CUES

16. The limit of tolerable aircraft response was derived by the pilot through a mental integration of the cues experienced and his knowledge of the recovery characteristics of the aircraft. The recovery cues were acceleration and rate of motion about the roll axis, the rotor rpm decay rate, and the expected recovery characteristics of the aircraft. The derived limits were a maximum bank attitude and a minimum rotor rpm. The oscillograph and pilot comment data indicate that the pilot limits for this test were approximately 50 degrees of bank attitude and 250 rotor rpm. The aircraft motions were determined primarily from visual cues obtained outside the cockpit. Kinesthetic cues were apparent but not as strong and clear as the visual cues regarding the rate of roll and the acceleration. The rotor rpm decay rate cues were obtained visually from a 3-inch sensitive rotor tachometer, located in the upper left portion of the instrument panel. Rotor rpm decay information was much easier to read accurately from this test instrument than the standard dual engine/rotor tachometer which was installed in the normal position. The artificial horizon was the initial production type. The current production M55 (Lear Seigler) attitude indicating system, MWO-1520-221-30/19, was not installed on the test aircraft. This attitude indicating system was installed in the aircraft evaluated during previous tests (refs 6 and 7, app I). The installed artificial horizon was unreliable for determining the aircraft attitude or rate of roll. Audio cues (from the engine, transmission, rotor and low rotor rpm warning) were apparent to the pilot but were not strong, clear, or, most important, quick enough to be useful in the short time interval between simulated engine failure and initiation of recovery. For night, limited visibility, or full instrument flight conditions, a much longer pilot recognition and reaction time must be possible since the primary cue, reference to the visual horizon, is not available. Reliable, accurate attitude instruments and clearer, quicker audio and visual cues are required for safe autorotational entries under these conditions.

RECOVERY TECHNIQUE

17. The sequence of pilot actions required to recover full control of the aircraft and enter stabilized autorotation was the same for all entry conditions evaluated. The first required action was to control the aircraft attitude; right lateral cyclic was applied to stop the left roll rate, aft cyclic to initiate a nose-up pitch rate and directional control to check the yaw and reduce the sideslip. The nose-up pitch rate was essential to stop the rotor rpm decay and reduce the high rate of descent. The cyclic inputs were the critical action and determined the maximum control delay time. When this action was accomplished smoothly and properly, aircraft control was fully retained by the pilot. The remaining actions to enter steady autorotation were less critical, and a wide variation of techniques could be employed. The nose-up pitch rate and the collective were smoothly adjusted to achieve and maintain the desired airspeed, 75 to 85 KIAS, and rotor rpm, 300 to 320 rpm. The time interval for accomplishing these actions varied from 1 to 5 seconds after the initial cyclic movements. The rate and size of recovery control inputs used were limited by cyclic feedback. Abrupt, large cyclic or pedal

inputs complicate and delay the recovery to controlled flight. Abrupt lowering of the collective before a substantial nose-up pitch rate and positive load factor is achieved causes large excursions of the main rotor flapping angle. Exceeding the maximum flapping angle would result in hub-mast contact with potentially disastrous results.

MILITARY SPECIFICATION COMPLIANCE

18. The military specification for helicopter flying qualities, MIL-H-8501A (ref 9, app I), was used in an attempt to determine the limits of acceptable flying qualities for autorotation entry maneuvers. Paragraphs 3.5.5 and 3.5.5.1 of MIL-H-8501A do not specify the delay required prior to movement of the cyclic and directional controls. If the 2-second delay specified in MIL-H-8501A for the collective control is applied to the other controls, the AH-1G helicopter does not meet the 10-degree attitude change throughout most of the present airspeed/engine torque operating envelope. Zero time delay on the cyclic is impractical and cannot be considered for determining acceptability. Some pilot recognition and reaction time must be specified with consideration given to the type and adequacy of the cues and to the type and number of pilot actions required. MIL-H-8501A is inadequate for determining the acceptability of any helicopter flying qualities during entry to autorotational flight.

19. The AH-1G characteristics were also compared to the proposed military specification, *Flying Qualities of Piloted V/STOL Aircraft* (ref 10, app I). The requirements of paragraph 3.8.11.1 state that attitude changes greater than 20 degrees in 2 seconds shall be considered excessive and that a control delay of 2 seconds is desirable. These requirements were not met at any of the airspeeds tested with an engine torque greater than 27 psi. For all those conditions, the roll attitude change was greater than 20 degrees in 2 seconds (fig. 19, app IV). To comply with the provisions of this specification would severely restrict the airspeed/engine torque envelope, so much so as to render the AH-1G unsuitable for the attack mission. Extensive experience in actual operations with this aircraft have shown that such restrictions would be difficult to justify. However, some design and evaluation criteria regarding the limits of acceptable flying qualities for this emergency maneuver are considered essential. The proposed limits are highly desired, but they probably should be varied according to the classification of aircraft, the probability of failure, and the use spectrum. Acceptability based on an attitude change in a defined time interval with all controls fixed is supported by the findings of this test. Since this test was conducted on a highly maneuverable, attack aircraft, the derived limits of attitude (bank) change for a reasonable delay interval are considerably greater than those specified as excessive. The specified value may be suitable for another class of aircraft, such as large, heavy, low-to-medium-maneuverability aircraft (class III). Further study should be conducted to determine the appropriate limits for each class of aircraft.

RESPONSE LIMITS

20. The limits of acceptable attitude change, rotor speed decay and pilot recognition and reaction time for the AH-1G have been qualitatively developed. These limits are based on the results of these tests and the considerable operational and training data available for this aircraft. The limits of acceptability were strongly influenced by the fact that the high airspeed, high engine torque conditions cannot be sustained for more than a few seconds. Also, combinations of engine torques greater than 35 psi and airspeeds greater than 140 KCAS are rarely required during tactical maneuvers. Additionally, it is assumed that the pilot will be actively controlling the aircraft and concentrating largely on the aircraft's flight path and attitude during the high-airspeed segment of a dive.

21. Roll attitude changes of approximately 40 degrees in 2 seconds were determined to be the acceptable limit for this aircraft. Since no significant pitch or yaw attitude changes were experienced, this value (40-degree attitude change in 2 seconds) is considered a limit for the sum of the attitude changes in all axes. A control delay time of 1.5 seconds for pilot recognition and reaction was determined to be the minimum acceptable for daytime, visual flight conditions. Figure 26 summarizes the delay times achieved at all combinations of entry engine torque and airspeed in the form of a cross plot from figures 24 and 25. All curves were extrapolated to 180 KCAS from 170 KCAS data. The curves for 1.75 and 1.5 seconds are extrapolated data for airspeeds less than 140 KCAS. For airspeeds greater than 140 KCAS, the roll attitude change of approximately 40 degrees in 2 seconds for control delays of 1.5 seconds was acceptable. The transient minimum rotor speeds for these conditions are acceptable.

22. Maximum delay times were between 1.2 and 1.5 seconds at engine torque settings greater than 42 psi for airspeeds between 120 and 140 KCAS. The level flight path provides a less difficult recovery situation than a high rate-of-descent dive. The aircraft response was slightly less severe for given torque settings at these airspeeds than at the higher dive airspeeds. Although more diversion of attention is probable in cruise flight, the visual and kinesthetic cues are considered adequate to cause proper pilot action in the time interval available.

23. The minimum acceptable pilot recognition and reaction time for flight under conditions where external visual cues are limited or nonexistent was not specifically determined during these tests. This should be determined during an instrument flying qualities evaluation, which has been previously recommended for the AH-1G (refs 1 through 6, app I). Based on the results of these and previous tests of the AH-1G, the night and limited-visibility normal operational envelope should be limited to those conditions where a 2-second control delay was achieved.

24. The engine torque/airspeed envelope recommended is presented in figure 27, appendix IV. The recommended operating envelopes are presented for both daytime, visual flight conditions and for night and/or limited-visibility flight conditions with the special caution area between 120 and 140 KCAS. The maximum airspeed for flight under instrument flying conditions must be determined

by further testing. The engine failure cues and the aircraft characteristics following engine failure are considered such that airspeeds greater than 150 KCAS are not recommended for flight at night or in limited-visibility conditions. Other handling problems may further limit the operating envelope during instrument flight conditions. Section IV of the operator's manual (ref 11, app I) should be revised to incorporate the preceding information and figure 27, appendix IV.

SCAS OFF

25. The tests to evaluate the response differences with SCAS OFF and ON were conducted at 100 KCAS in the light weight, aft cg, clean wing configuration. The results are presented in figures 3, 7, 11, 14, 17 and 20, appendix IV. The aircraft reactions with the SCAS ON at 100 KCAS were moderate, and the entry into autorotation was easily accomplished. Control delay times were 2 seconds or greater for most entry engine torque values tested. With SCAS OFF, the response was severe and the highest engine torque setting evaluated was 40 psi where the control delay was less than 1 second. The large increase in roll rate and acceleration with the SCAS OFF is shown in figure 7. Due to the severity of the response at this moderate airspeed and the large difference in characteristics with SCAS ON and OFF, no tests were made at higher airspeeds. It is recommended that SCAS OFF flight be limited to 100 KCAS and that no operations other than return-to-base or ferry flights be conducted with either the lateral or directional SCAS channels inoperative. The lack of motion about the pitch axis following sudden engine failure makes the pitch channel less critical, and no limitations are recommended when the pitch channel is inoperative. Additionally, high power settings should be avoided when operating with the lateral and directional SCAS channels inoperative because of the Dutch-roll instability at airspeeds between 60 and 100 KCAS (refs 3 and 5, app I).

CONCLUSIONS

26. The response of the AH-1G helicopter following sudden engine failure is unacceptably severe at high airspeeds with maximum engine torque applied (para 7).
27. The severity of the aircraft response is primarily a function of the engine torque at the time of failure with some airspeed effect (para 15).
28. The response of the AH-1G to sudden engine failures at reduced engine torque values provides adequate visual and kinesthetic cues under daytime, visual flight conditions to cause the pilot to take prompt, natural actions to effect recovery and entry into autorotation (para 16).
29. The cues available following sudden engine failure under night or limited-visibility flight conditions are insufficient, and normal operations should be limited to engine torque/airspeed conditions which produce a less severe response than acceptable for daytime, visual flight operations (paras 16, 23 and 24).
30. The limits of acceptable response to sudden engine failure for the AH-1G helicopter were determined to be a bank attitude of approximately 50 degrees and a minimum rotor speed of 250 rpm (para 16).
31. The time for initiation of recovery action was determined by the pilot by considering the aircraft roll rate, roll acceleration, rotor rpm decay rate, and recovery characteristics (para 17).
32. The present and proposed military specifications for flying qualities of helicopters following sudden engine failure are inadequate and should be revised (paras 18 and 19).
33. The daytime, visual flight conditions operating envelope of the AH-1G should be limited to those combinations of engine torque and airspeed where a 1.5-second control delay was achieved and the bank attitude change was not more than 40 degrees in 2 seconds (para 21).
34. The night and limited-visibility operating envelope for the AH-1G should be limited, in the interim, to those conditions where a 2-second control delay was achieved (para 23).
35. Clear and quicker audio and visual cues are required for safe autorotational entries under all authorized flight conditions (para 16).
36. Further testing of the AH-1G under night and limited-visibility flight conditions is required to define the operating envelope (para 24).

37. The aircraft response following sudden engine failure with SCAS OFF is very severe, and safe entry into autorotation is doubtful at airspeeds greater than 100 KCAS with high engine torque settings (para 25).

RECOMMENDATIONS

38. The AH-1G should not be normally operated at combinations of engine torque and airspeed greater than those shown as safe in figure 27, appendix IV.
39. Figure 27, appendix IV, showing the recommended airspeed/engine torque limits for normal operations, should be incorporated into the AH-1G operator's manual, TM 55-1520-221-10 (ref 11, app I).
40. Paragraphs 4-16 and 4-17 of the AH-1G operator's manual should be revised to incorporate the description of the aircraft response, pilot cues, and recovery technique as presented in this report.
41. The AH-1G should be limited to airspeeds less than 100 KCAS when the lateral or directional SCAS channels are inoperative. Only return-to-base or ferry flights should be attempted in this situation.
42. Further tests and studies should be accomplished to define specification criteria for the flight characteristics of all classes and types of V/STOL aircraft following sudden engine failure.
43. Further tests should be conducted on the AH-1G helicopter to define the safe operating envelope under night and limited-visibility conditions.
44. Clearer, quicker and more reliable audio and visual warnings and cues should be developed for this aircraft.

APPENDIX I. REFERENCES

1. Final Report, US Army Aviation Test Activity (USAAVNTA), Project No. 66-06, *Engineering Flight Test of the AH-1G Helicopter (HueyCobra), Phase B, Part 1*, January 1968.
2. Final Report, USAASTA, Project No. 66-06, *Engineering Flight Test of the AH-1G Helicopter (HueyCobra), Phase B, Part 2*, May 1969.
3. Final Report, USAASTA, Project No. 66-06, *Engineering Flight Test of the AH-1G Helicopter (HueyCobra), Phase D, Part 1, Handling Qualities*, December 1970.
4. Final Report, USAASTA, Project No. 69-01, *Airworthiness and Flight Characteristics Test, AH-1G Helicopter with Stabilized Night Sight (SNS), Phase I*, December 1969.
5. Final Report, USAASTA, Project No. 69-01, *Airworthiness and Flight Characteristics Test, AH-1G Helicopter with Stabilized Night Sight (SNS), Phase II*, August 1970.
6. Letter, USAASTA, 25 September 1970, subject: Final Report, Qualitative Airworthiness Assessment of AH-1G HueyCobra Night Fire Control System (CONFICS), Project No. 70-08.
7. Final Report, USAASTA, Project No. 70-10, *Airworthiness and Flight Characteristics Test (Limited), AH-1G Helicopter (SMASH)*, October 1970.
8. Letter, with inclosure, USAAVSCOM, 4 August 1970, subject: Test Directive, USAAVSCOM, No. 70-25, AH-1G Reduced Torque Autorotations.
9. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For*, 7 September 1961, Amended 3 April 1962.
10. Military Specification, MIL-F-(Proposed), *Flying Qualities of Piloted V/STOL Aircraft*, July 1970.
11. Technical Manual, TM 55-1520-221-10, *Operator's Manual, Army Model AH-1G Helicopter*, April 1969.

APPENDIX II. BASIC AIRCRAFT INFORMATION AND OPERATING LIMITS

AIRFRAME

Rotor System

1. The 540 "door hinge" main rotor assembly is a two-bladed semirigid, underslung feathering-axis type rotor. The assembly consists of two all-metal blades, blade grips, yoke extensions, yoke trunnion, and rotating controls. Control horns for cyclic and collective control input are mounted on the trailing edge of the blade grip. Trunnion bearings permit rotor flapping. The blade grip-to-yoke extension bearings permit cyclic and collective pitch action.

Tail Rotor

2. The tail rotor is a two-bladed, delta-hinge type employing precone and underslung. The blade and yoke assembly is mounted to the tail rotor shaft by means of a delta-hinge trunnion. Blade pitch angle is varied by movement of the tail rotor control pedals. Power to drive the tail rotor is supplied by a takeoff on the lower end of the main transmission.

Transmission System

3. The transmission is mounted forward of the engine and coupled to the engine by a short drive shaft. The transmission is a reduction gearbox which transmits engine power at reduced rpm to the main and tail rotors by means of a two-stage planetary gear train. The transmission incorporates a free-wheeling clutch unit at the input drive. This provides a disconnect from the engine in case of a power failure to allow the aircraft to make an autorotational landing.

Synchronized Elevator

4. The synchronized elevator, which has an inverted airfoil section, is located near the aft end of the tail boom and is connected by control tubes and mechanical linkage to the fore and aft cyclic control system. Fore and aft movements of the cyclic control stick produce a change in the synchronized elevator attitude.

Control Systems

5. A dual hydraulic control system is provided for the cyclic and collective controls. The directional controls are powered by a single servo cylinder which is operated by system number 1. The hydraulic system consists of two hydraulic pumps, two reservoirs, relief valves, shut-off valves, pressure warning lights, lines, fittings, and manual dual-tandem servo actuators incorporating irreversible valves. Tandem power cylinders incorporating closed-center four-way manual servo valves and irreversible valves are provided in the lateral, fore and aft cyclic and collective

control system. A single power cylinder incorporating a closed-center four-way manual servo valve is provided in the directional control system. The cylinders contain a straight-through mechanical linkage.

Force Trim

6. Magnetic brake and force gradient devices are incorporated in the cyclic control and directional pedal controls. These devices are installed in the flight control system between the cyclic stick and the hydraulic power cylinders and between the directional pedals and the hydraulic power cylinder. The force trim control can be turned off by depressing the left button on the top of the cyclic stick. The gradient is accomplished by springs and magnetic brake release assemblies which enable the pilot to trim the controls as desired.

Cyclic Control Stick

7. The pilot and gunner cyclic stick grips each have a force trim switch and a SCAS release switch. The pilot cyclic stick has a built-in operating friction. The cyclic control movements are transmitted directly to the swash plate. The fore and aft cyclic control linkage is routed from the cyclic stick through the SCAS actuator, to the dual boost hydraulic actuator, and then to the right horn of the fixed swash plate ring. The lateral cyclic is similarly routed to the left horn.

Collective Pitch Control

8. The collective pitch control is located to the left of the pilot and is used to control the vertical mode of flight. Operating friction can be induced into the control lever by hand-tightening the friction adjuster. The pilot and gunner collective pitch controls have a rotating grip-type throttle.

Tail Rotor Pitch Control Pedals

9. Tail rotor pitch control pedals alter the pitch of the tail rotor blades and thereby provide the means for directional control. The force trim system is connected to the directional controls and is operated by the force trim switch on the cyclic control grip.

Stability and Control Augmentation System

10. The SCAS is a three-axis, limited-authority, rate-referenced stability augmentation system. It includes an electrical input which augments the pilot mechanical control input. This system permits separate consideration of airframe displacements caused by external disturbances from displacements caused by pilot input. The SCAS is integrated into the fore, aft, lateral and directional flight controls to improve the stability and handling qualities of the helicopter. The system consists of electro-hydraulic servo actuators, control motion transducers, a sensor/amplifier unit and a control panel. The servo actuator movements are not felt by the pilot. The actuators are limited to a 25-percent authority and will center and lock in case of an electrical and/or a hydraulic failure.

ENGINE

Engine Description

11. The T53-L-13 engine, rated at 1400 shp, is a successor to the T53-L-11 engine. The additional power has been achieved with no change in the basic T53-L-11 engine envelope mounting and connection points and with a 6-percent increase in basic engine weight.

12. The performance gain is accomplished thermodynamically by the mechanical integration of a modified axial compressor, a two-stage compressor turbine and a two-stage power turbine into the T53-L-11 engine configuration.

13. Replacement of the first two compressor stators and changing of the first two stages of compressor rotor blades and discs results in an approximate 20-percent increase in mass air flow through the engine. This is accomplished without the use of inlet guide vanes.

14. An inlet flow fence, located on the outer wall of the inlet housing in the area of the previously used inlet guide vanes, provides the desired inlet conditions for the transonic compression during acceleration at low speeds. At compressor speeds up to 70 percent, the fence is in the extended position. Above 70 percent, the flow fence is retracted into the outer wall of the inlet housing. Similar to a piston ring, the circumference of the flow fence is changed by the action of a piston actuator powered by compressor discharge pressure.

15. The specification for this engine allows the use of JP-4 or JP-5 fuel for satisfactory operation throughout the engine's operating envelope. During this program, JP-4 fuel was used.

Engine Power Control System

16. The fuel control for the T53-L-13 engine is a hydro-mechanical type of fuel control. It consists of the following main units:

- a. Dual-element fuel pump.
- b. Gas producer speed governor.
- c. Power turbine speed topping governor.
- d. Acceleration and deceleration control.
- e. Fuel shut-off valve.
- f. Transient air bleed control.

17. An air bleed control is incorporated within the fuel control to provide for opening and closing the compressor interstage air bleed in response to the following signals present in the power control:

- a. Gas producer speed.
- b. Compressor inlet air temperature.
- c. Fuel flow.

18. The fuel control is designed to be operated either automatically or in an emergency mode. In the emergency position, fuel flow is terminated to the main metering valve and is routed to the manual (emergency) metering and dump valve assembly. While in the emergency mode, fuel flow to the engine is controlled by the position of the manual metering valve which is connected directly to the power control (twist grip). During the emergency operation, there is no automatic control of fuel flow during acceleration and deceleration; thus, engine acceleration and exhaust gas temperature (EGT) must be pilot monitored.

BASIC AIRCRAFT INFORMATION

Airframe Data

Overall length (rotor turning)	637.2 in.
Overall width (rotor trailing)	124.0 in.
Centerline of main rotor to centerline of tail rotor	320.7 in.
Centerline of main rotor to elevator hinge line	198.6 in.
Elevator area (total)	15.2 sq ft
Elevator area (both panels)	10.9 sq ft
Elevator airfoil section	Inverted Clark Y
Vertical stabilizer area	18.5 sq ft
Vertical stabilizer airfoil section	Special camber
Vertical stabilizer aerodynamic center	Fuselage station (FS) 499.0

Wing area:

Total 27.8 sq ft

Outboard of butt line (BL) 18.0
(both sides) 18.5 sq ft

Wing span 10.33 ft

Wing airfoil section:

Root NACA 0030

Tip NACA 0024

Wing angle of incidence 14 deg

Main Rotor Data

Number of blades 2

Diameter 44 ft

Disc area 1520.5 sq ft

Blade chord 27 in.

Rotor solidity 0.0651

Blade area (both blades) 99 sq ft

Blade airfoil 9.33 percent symm
special section

Linear blade twist -0.455 deg/ft

Hub precone angle 2.75 deg

Rotor inertia 2900 slug-ft²

Antitorque Rotor Data

Number of blades 2

Diameter 8.5 ft

Disc area 56.74 sq ft

Directional SCAS authority:

± 12.5 percent or ± 0.88 inch of directional
(pedal) control displacement

OPERATING LIMITATIONS

Limit Airspeed

Any configuration with XM159 rocket pods:

180 KCAS below a 3000-foot H_D ; decrease 8 KCAS per 1000 feet above
3000 feet

All other configurations:

190 KCAS below a 4000-foot H_D ; decrease 8 KCAS per 1000 feet above
4000 feet

Gross-Weight/Center-of-Gravity Envelope

Forward cg limit:

Below 7000 pounds, FS 190.0; linear increase to FS 192.1 at 9500 pounds

Aft cg limit:

Below 8270 pounds, FS 201.0; linear decrease to FS 200 at 9500 pounds

Sideslip Limits

Five degrees at 190 KCAS with linear increase to 20 degrees at 60 KCAS

Rotor and Engine Speed Limits (Steady State)

Power on:

Engine rpm	6400 to 6600
Rotor rpm	314 to 324

Power off:

Rotor rpm	294 to 339
Rotor rpm transient lower limit	250

Blade chord	8.41 in.
Rotor solidity	0.105
Blade airfoil	NACA 0010 modified
Blade twist	Zero deg

Transmission Drive System Ratios

Engine to main rotor	20.383:1.0
Engine to antitorque rotor	3.990:1.0
Engine to antitorque drive system	1.535:1.0

Test Aircraft (S/N 6615247) Control Displacements

Longitudinal cyclic control:

Full forward to full aft with SCAS nulled	9.07 in.
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Lateral cyclic control:

Full left to full right with SCAS nulled	10.00 in.
--	-----------

Directional (pedal) control:

Full left to full right with SCAS nulled	7.07 in.
--	----------

Collective control:

Full up to full down with SCAS nulled	9.30 in.
---------------------------------------	----------

Test Aircraft (S/N 6615247) SCAS Authority

Longitudinal SCAS authority:

±12.5 percent or ±1.13 inches of longitudinal cyclic control displacement

Lateral SCAS authority:

±12.5 percent or ±1.25 inches of lateral cyclic control displacement

Power on during dives and maneuvers:

Rotor rpm	314 to 324
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Temperature and Pressure Limits

Engine oil temperature	93°C
Transmission oil temperature	110°C
Engine oil pressure	25 to 100 psi
Transmission oil pressure	30 to 70 psi
Fuel pressure	5 to 20 psi

T53-L-13 Engine Limits

Normal rated EGT (maximum continuous)	625°C
Military rated EGT (30-minute limit)	645°C
Starting and acceleration EGT (5-second limit)	675°C
Maximum EGT for starting and acceleration	760°C
Torque pressure limit	50 psi

Test Aircraft (S/N 6615247) SCAS Authority

Longitudinal SCAS authority:

±12.5 percent or ±1.13 inches of longitudinal
cyclic control displacement

Lateral SCAS authority:

±12.5 percent or ±1.25 inches of lateral
cyclic control displacement

Directional SCAS authority:

±12.5 percent or ±0.88 inch of directional
(pedal) control displacement

APPENDIX III. TEST INSTRUMENTATION

USA S/N 6615247

Flight test instrumentation was installed in the test helicopter during prior tests. This instrumentation provided data from three sources: pilot panel, copilot/engineer panel, and a 50-channel oscillograph. All instrumentation was calibrated. The flight test instrumentation was installed and maintained by the Instrumentation Branch, Logistics Division, USAASTA. The following test parameters were presented:

PILOT PANEL

(Standard system) airspeed
(Boom system) airspeed
(Boom system) altitude
Rate of climb
Gas producer speed
(Standard system) torque pressure
Exhaust gas temperature
Longitudinal control position
Lateral control position
Pedal control position
Collective control position
CG normal acceleration
Angle of sideslip

NOT REPRODUCIBLE

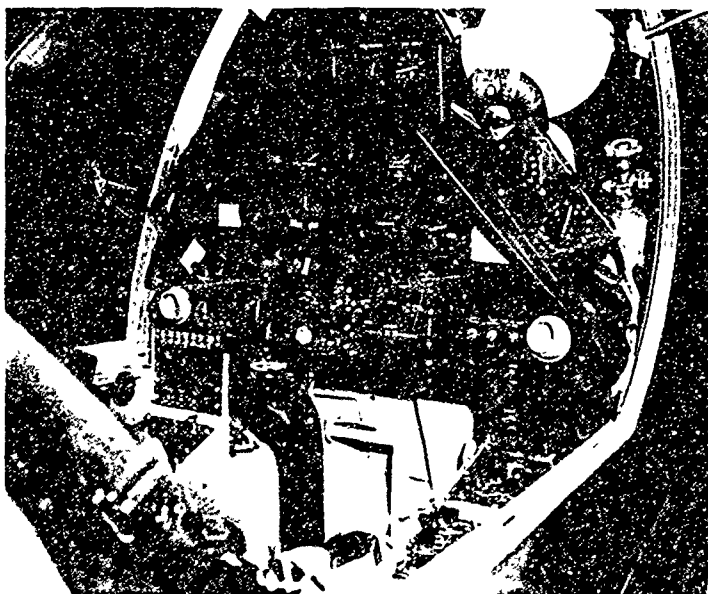


Photo 1. Pilot Panel.

ENGINEER PANEL

(Boom system) altitude
Outside air temperature
Rotor speed
Gas producer speed
Fuel used total
Torque pressure (high)
Torque pressure (low)
Exhaust gas temperature
Oscillograph correlation counter
Engine fuel flow
Voice tape recorder

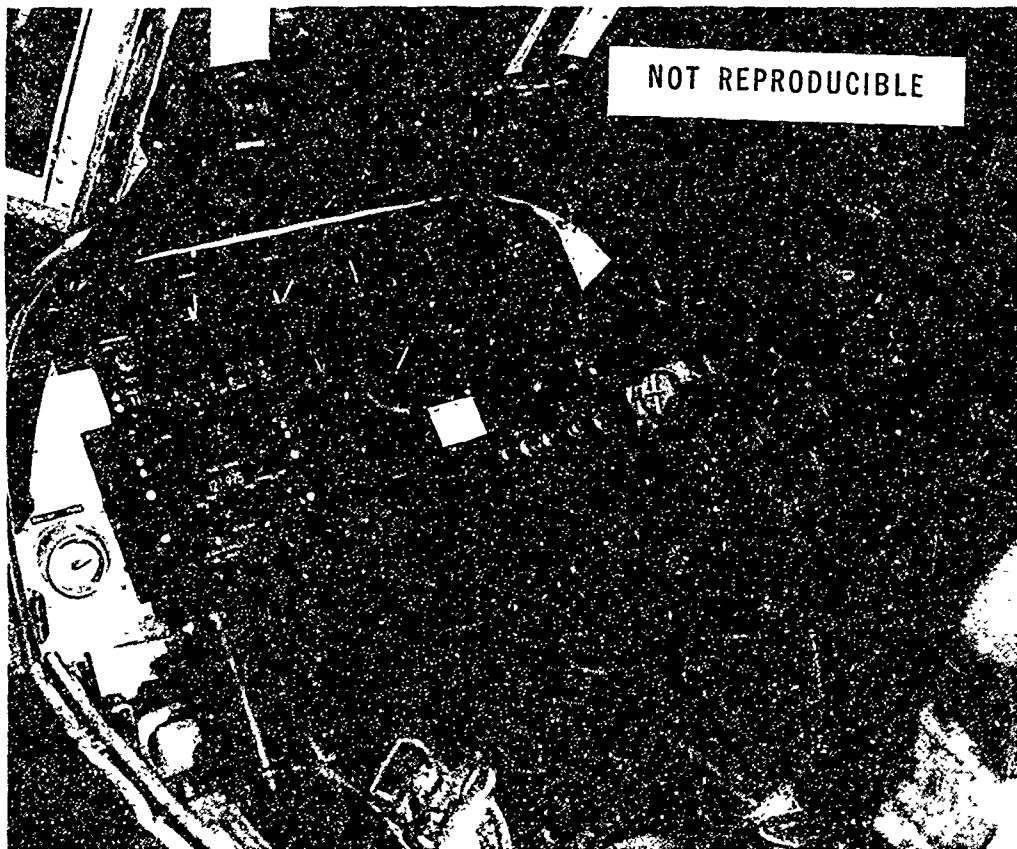


Photo 2. Copilot/Engineer Panel.

OSCILLOGRAPH

Longitudinal control position
Lateral control position
Directional control position
Collective control position
Pitch attitude
Roll attitude
Yaw attitude
Pitch rate
Roll rate
Yaw rate
CG normal acceleration
Angle of sideslip
Angle of attack
Linear rotor speed
Engine torque
Throttle position
Directional SCAS position
Lateral SCAS position
Longitudinal SCAS position
Engine rpm, N₂
Lateral linear acceleration
Longitudinal linear acceleration

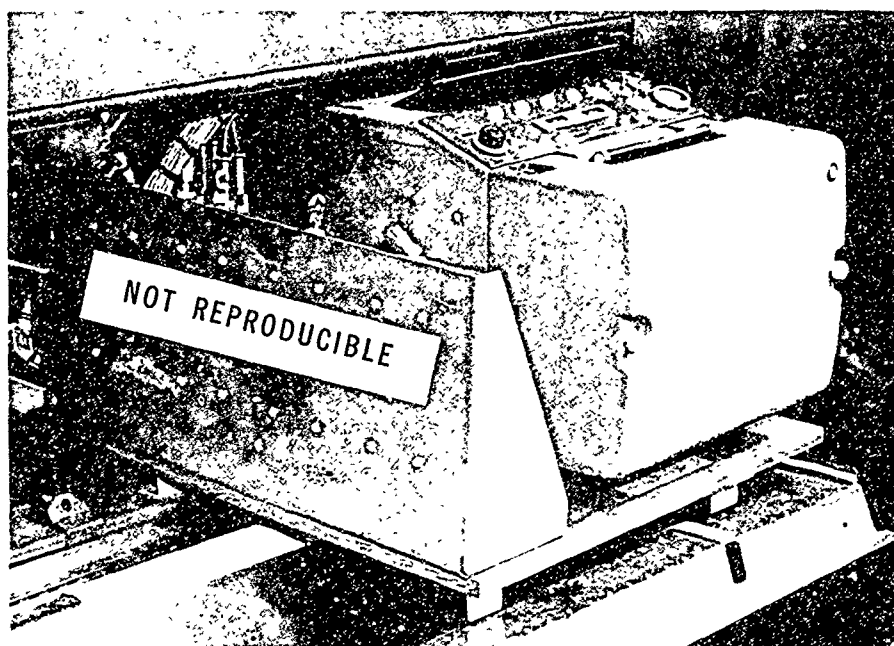


Photo 3. 24-Channel Oscillograph.

APPENDIX IV. TEST DATA

FIGURE 1
ROLL ATTITUDE VS ENTRY ENGINE TORQUE
 AH-1G USA WGI 5247

SYM.	DENSITY ALT	GRWT	LONG. CG.	ROTOR SPEED	SCAS	OAT	ARMAMENT
0	5000	7850	201.0 (AFT)	324	ON	14	CLEAN
	~FT.	~LB.	~IN.	~RPM		~°C	CONFIGURATION

NOTE 1. DASHED LINE DENOTES MAX ROLL ATTITUDE
 2. SOLID LINE DENOTES ROLL ATTITUDE @ RECOVERY INITIATION

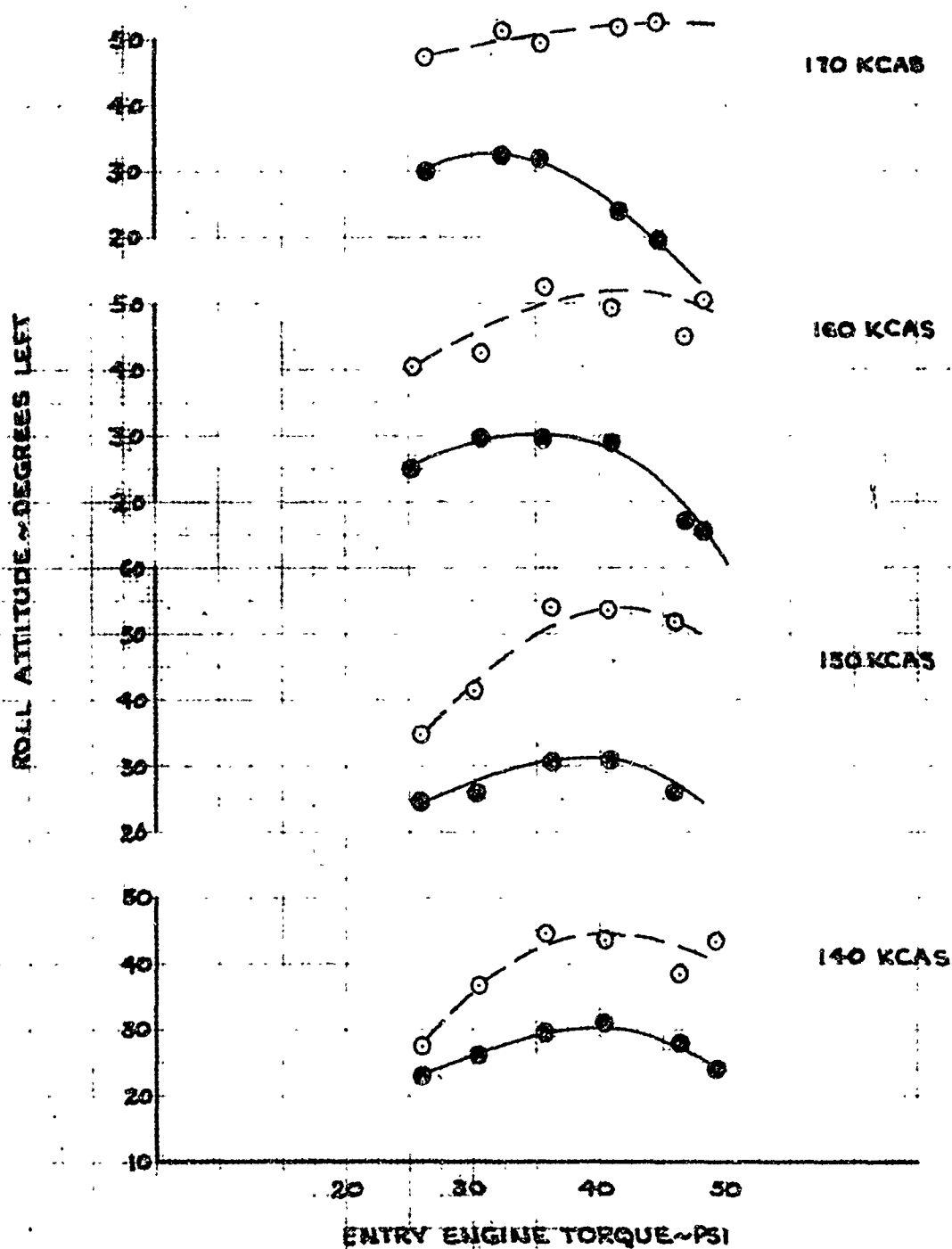


FIGURE 2
ROLL ATTITUDE VS ENTRY ENGINE TORQUE
AH-1G USA 54G15247

SYM	DENSITY	ALT.	GRWT.	LONG.C.G.	ROTOR SPEED	SCAS	OAT	ARMAMENT
Δ	$H_D \sim$ FT.		$\sim$ LB	$\sim$ IN.	$\sim$ RPM		$^{\circ}$ C	CONFIGURATION
	5000		9200	192.7 (FWD)	524	ON	6	HVY HOG

NOTE 1. BROKEN LINE DENOTES MAX ROLL ATTITUDE
 2. SOLID LINE DENOTES ROLL ATTITUDE @ RECOVERY INITIATION

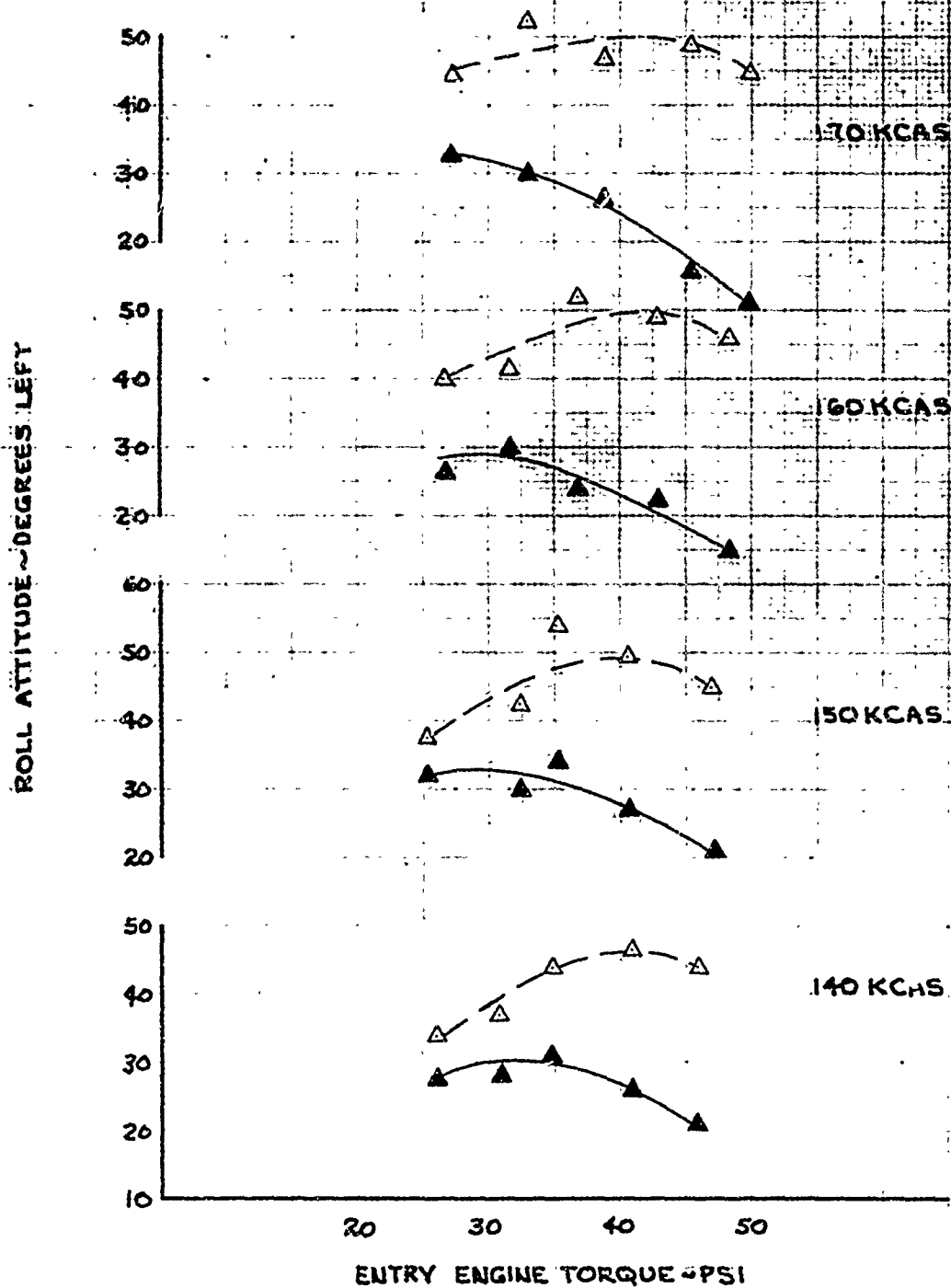


FIGURE 3
ROLL ATTITUDE VS ENTRY ENGINE TORQUE
 AH-1G USA 5/1615247

SYM	DENSITY H _D ~ FT	ALT. ~ FT	GRWT ~ LB	LONG.C.G. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
□	5000	7200	7200	201.0 (AFT)	324	ON	13	CLEAN
▽	5000	7200	7200	201.0 (AFT)	324	OFF	13	CLEAN

NOTE 1. BROKEN LINE DENOTES MAX. ROLL ATTITUDE
 2. SOLID LINE DENOTES ROLL ATTITUDE @ RECOVERY INITIATION

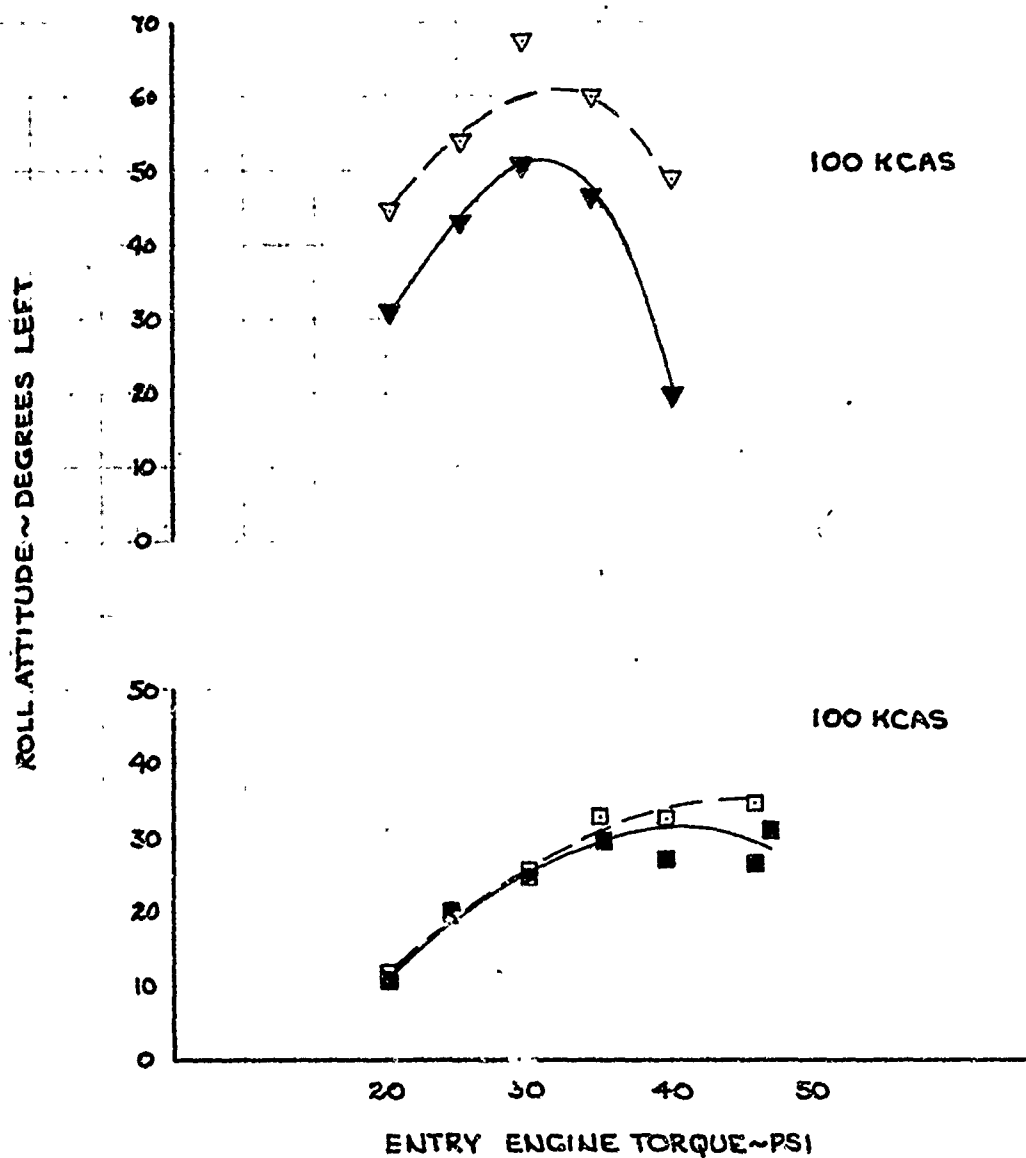


FIGURE 4
ROLL RATE AT ONE SECOND VS ENTRY ENGINE TORQUE
 AH-1G USA #1615247

SYM	DENSITY ALT. H _p ~ FT	GRWT ~ LB	LONG. CG. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT °C	ARMAMENT CONFIGURATION
O	5000	7350	201.0 (AFT)	324	ON	14	CLEAN
Δ	5000	9200	192.7 (FWD)	324	ON	6	HVY-HOG

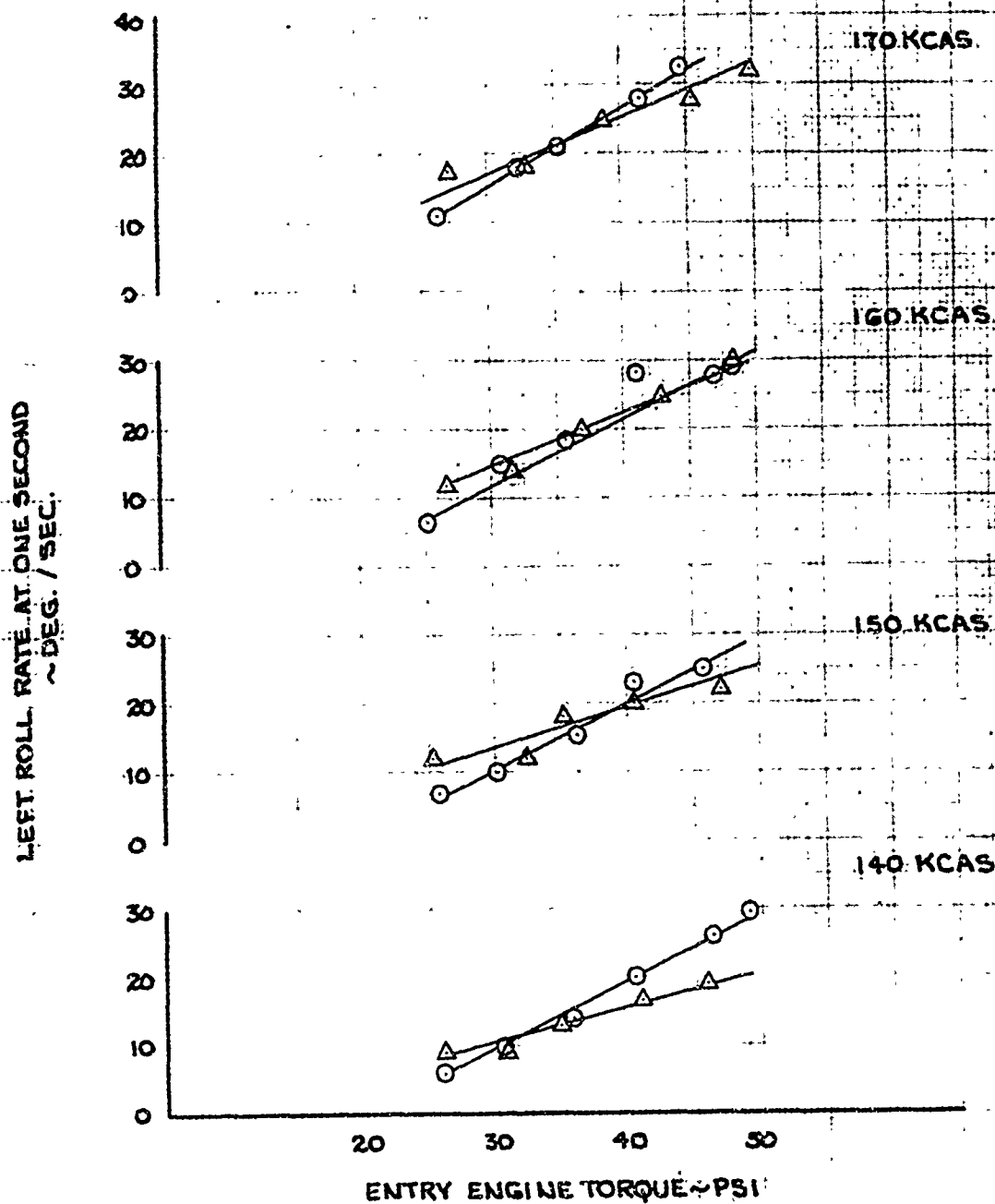


FIGURE 5
ROLL RATE VS ENTRY ENGINE TORQUE
 AH-1G USA 54G15247

SYM	DENSITY	ALT	GRWT	LONG.C.G	ROTOR SPEED	SCAS	OAT	ARMAMENT
○	H ₀ ~ FT	~ LB	~ IN.	~ RPM	ON	14	CLEAN	CONFIGURATION
○	5000	7350	201.0 (AFT)	324				

NOTE 1. BROKEN LINE DENOTES MAX. ROLL RATE
 2. SOLID LINE DENOTES ROLL RATE @ RECOVERY INITIATION

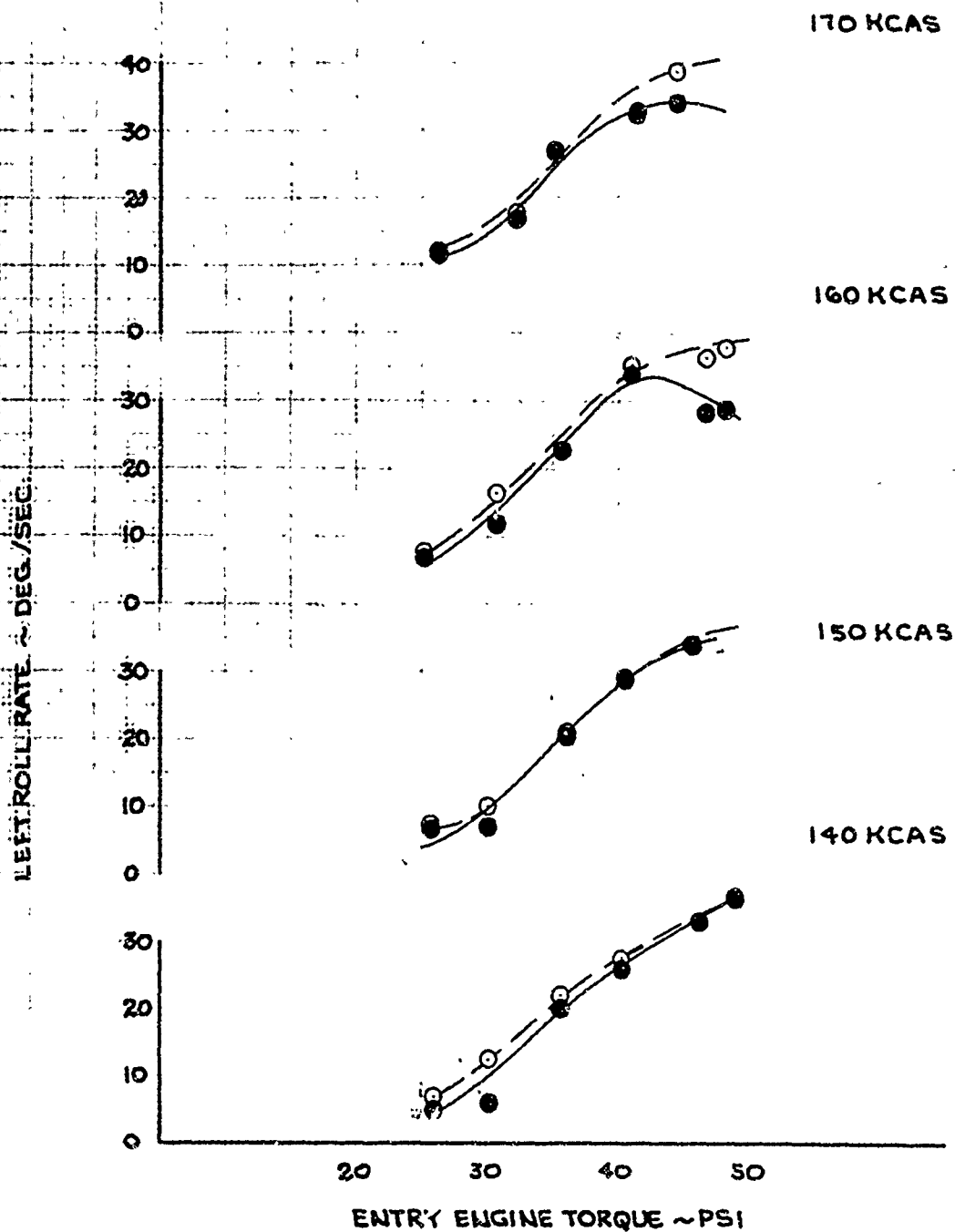


FIGURE 6 **ROLL RATE VS ENTRY ENGINE TORQUE** AH-1G USA %615247

SYM	DENSITY	ALT.	GRWT	LONG.C.G.	ROTOR SPEED	SCAS	OAT	ARMAMENT
	$H_D \sim$	~FT	~LB	~IN.	~RPM		°C	CONFIGURATION
Δ	5000		9200	192.7 (FWD)	324	ON	6	HVY. HOG

NOTE: 1. BROKEN LINE DENOTES MAX. ROLL RATE
 2. SOLID LINE DENOTES ROLL RATE @ RECOVERY INITIATION

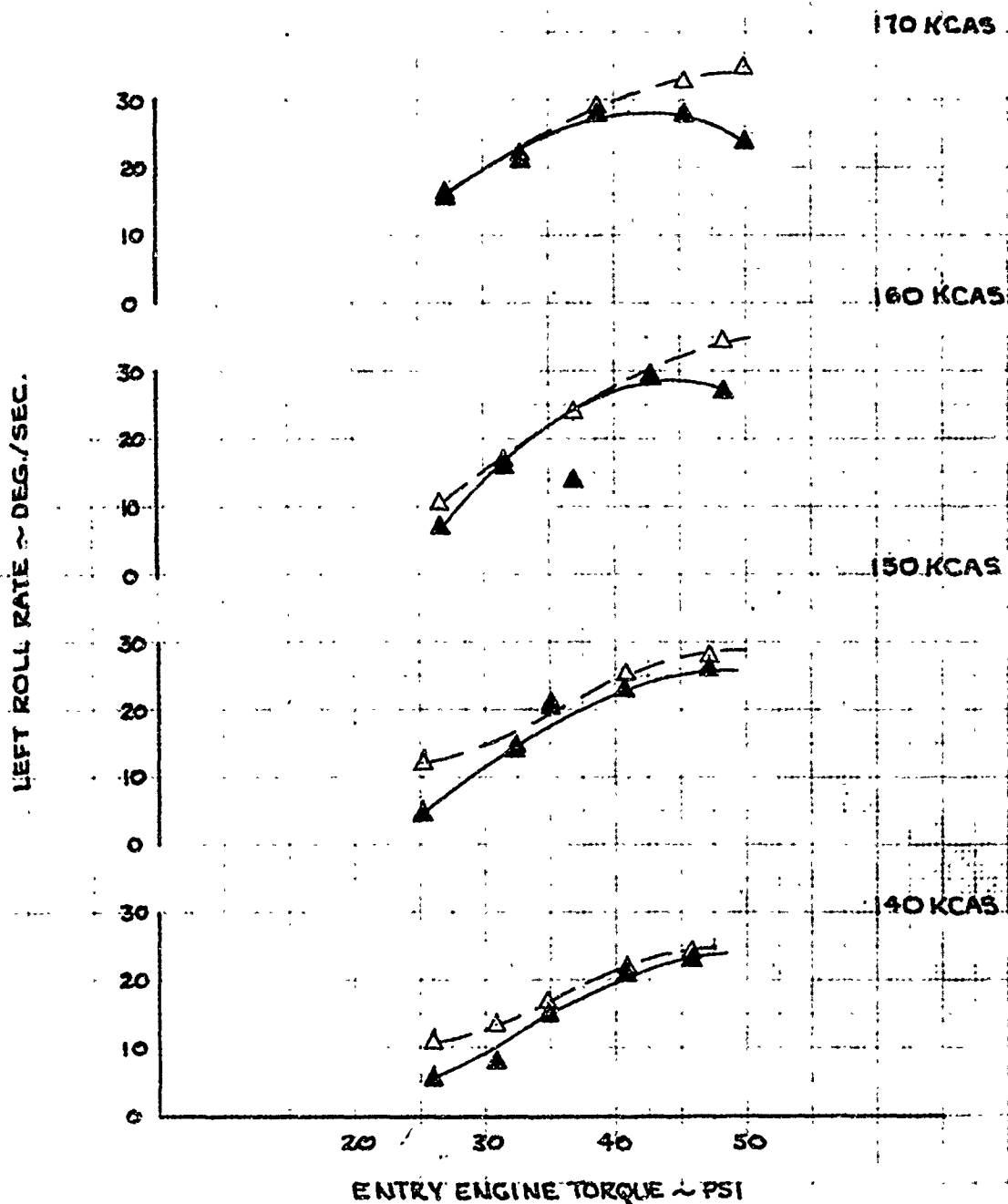


FIGURE 7
ROLL RATE AND ACCELERATION VS ENTRY ENGINE TORQUE

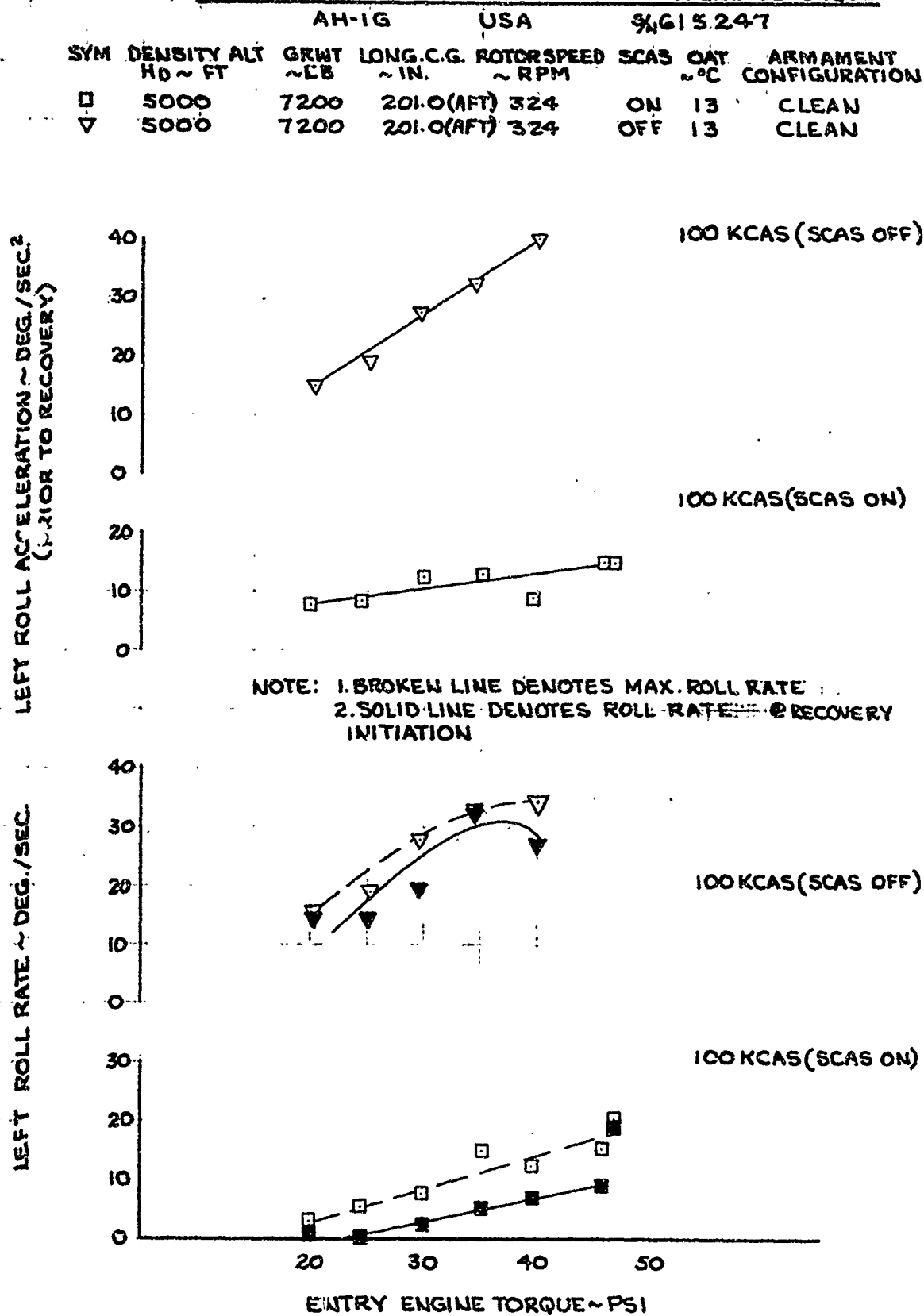


FIGURE 8
ROLL ACCELERATION VS ENTRY ENGINE TORQUE
AH-1G USA 1/4 G15247

SYM	DENSITY ALT. H ₀ ~ FT	GRWT ~ LB	LONG. C.G. ~ IN	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
O	5000	7350	201.0 (AFT)	324	ON	14	CLEAN
Δ	5000	9200	192.7 (FWD)	324	ON	6	HVY. HOG

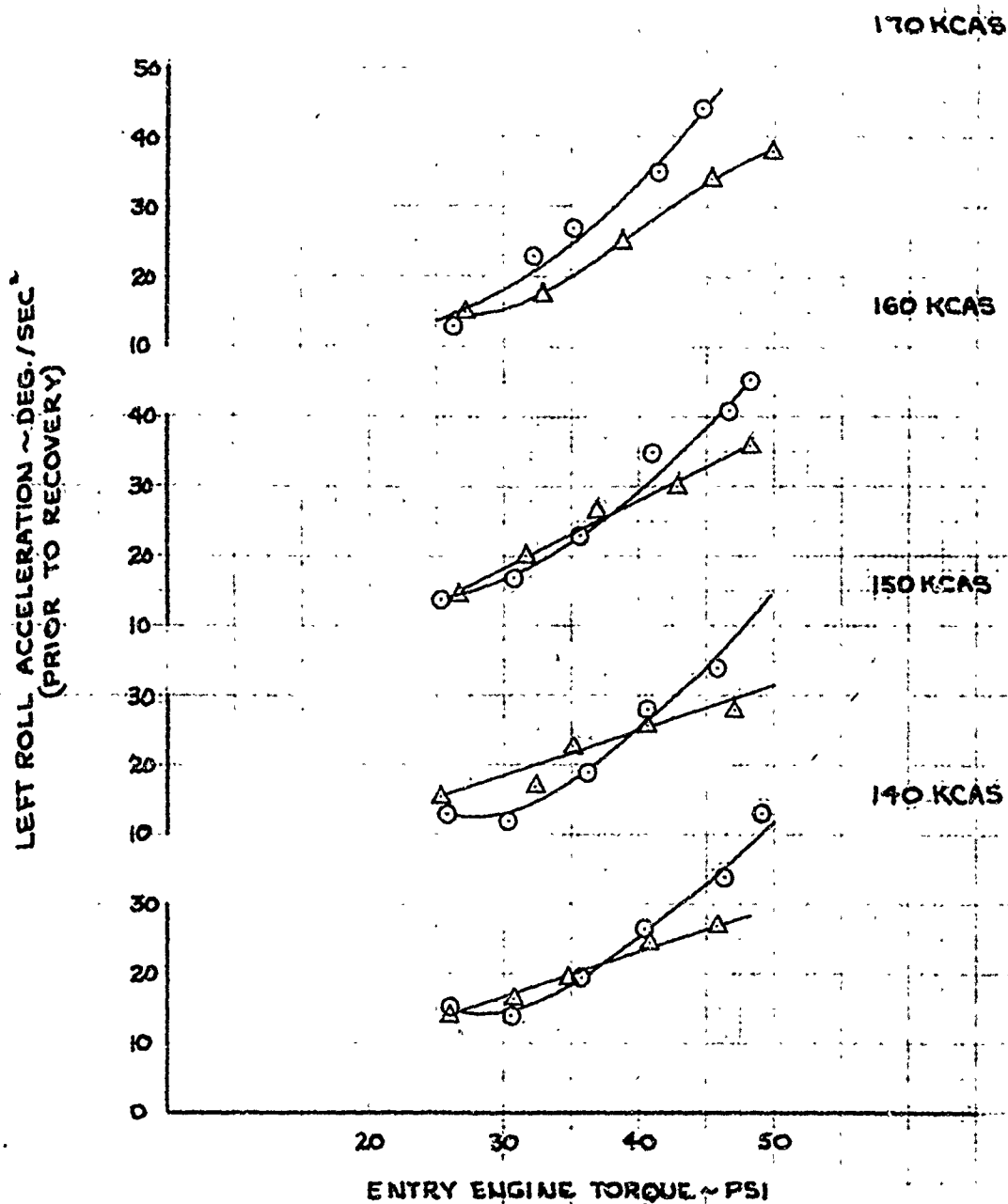


FIGURE 9
YAW RATE VS ENTRY ENGINE TORQUE
 AH-1G USA 5/615247

SYM.	DENSITY	ALT	GRWT	LONG.C.G.	ROTOR SPEED	SCAS	OAT	ARMAMENT
	H ₀ ~ FT		~ LB	~ IN.	~ RPM		~ °C	CONFIGURATION
0	5000		7350	201.0 (AFT)	324	ON	14	CLEAN

NOTE 1. BROKEN LINE DENOTES MAX. YAW RATE
 2. SOLID LINE DENOTES YAW RATE @ 1 SEC.

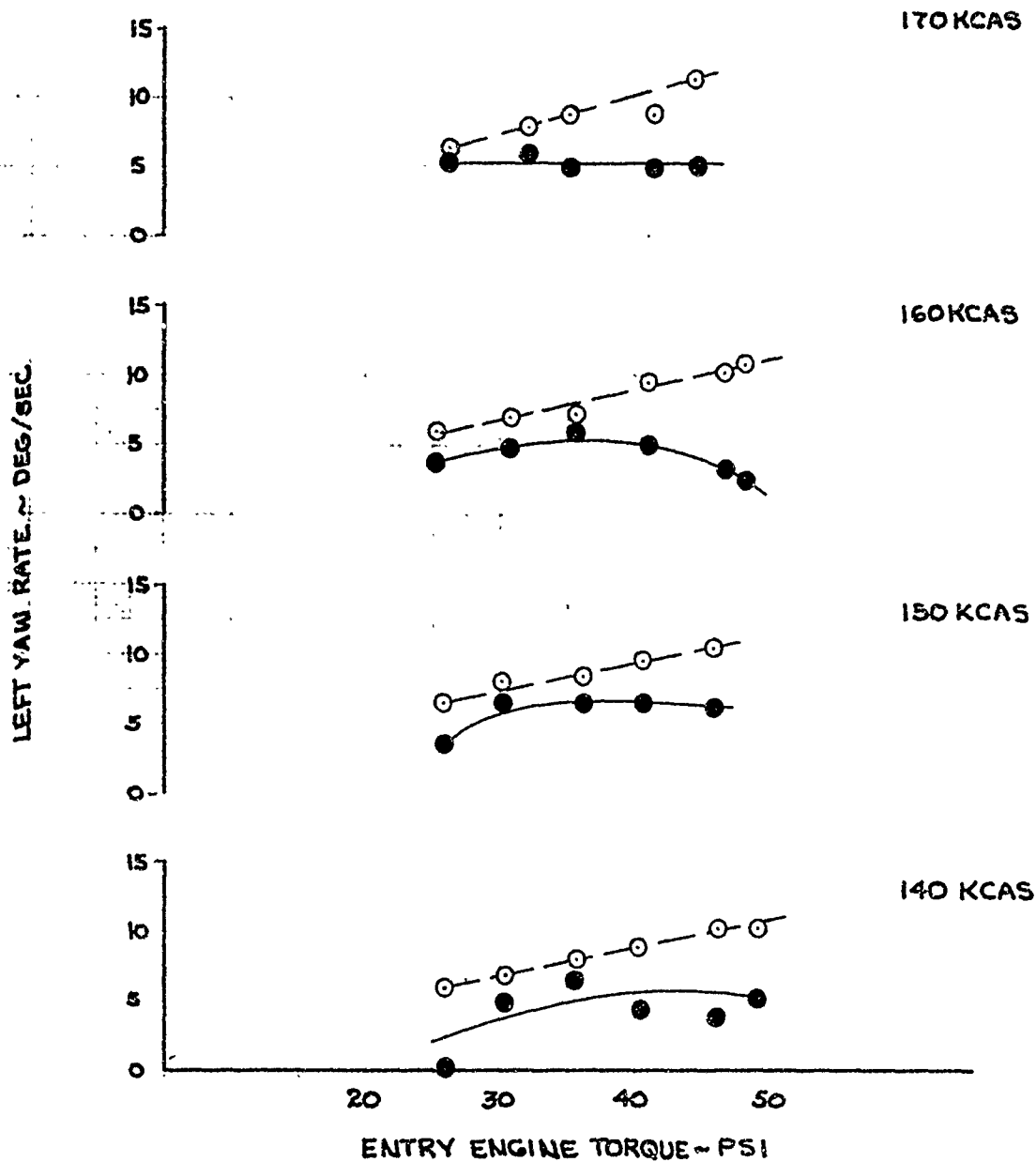


FIGURE 10
YAW RATE VS ENTRY ENGINE TORQUE
AH-1G USA S/N 15247

SYM	DENSITY	ALT.	GRWT	LONG. CG	ROTOR SPEED	SCAS	OAT	ARMAMENT
	H _D ~ FT		~ LB	~ IN	~ RPM		~ °C	CONFIGURATION
Δ	5000		9200	192.7 (FWD)	324	ON	6	HVY HOG

NOTE 1. BROKEN LINE DENOTES MAX. YAW RATE
 2. SOLID LINE DENOTES YAW RATE @ 1 SEC.

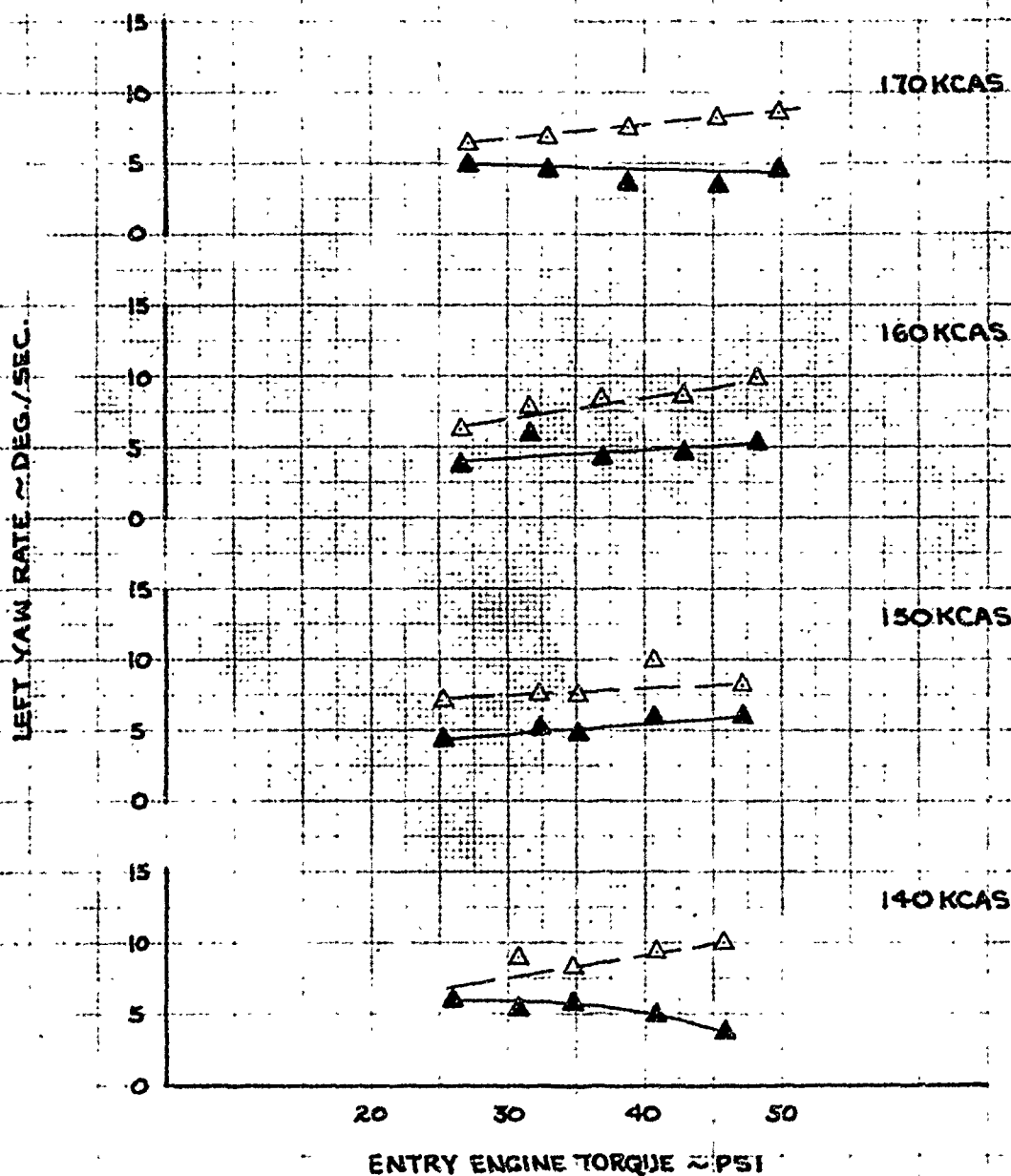


FIGURE 11

YAW RATE VS ENTRY ENGINE TORQUE

AH-1G USA 6/4615247

SYM	DENSITY H _D ~ FT	ALT ~ LB	GRWT ~ IN.	LONG. CG. ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
□	5000	7200	201.0(AFT)	324	ON	13	CLEAN
▽	5000	7200	201.0(AFT)	324	OFF	13	CLEAN

NOTE 1. BROKEN LINE DENOTES MAX. YAW RATE
2. SOLID LINE DENOTES YAW RATE @ 1 SEC.

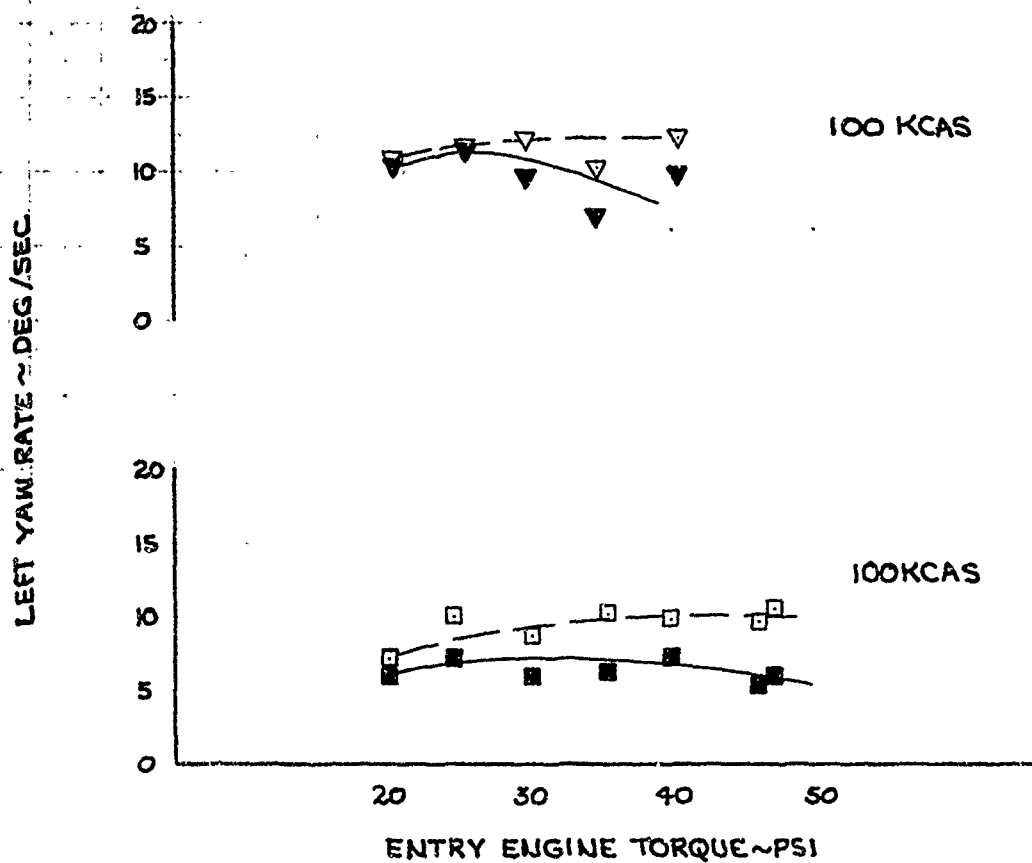


FIGURE 12
ANGLE OF SIDESLIP VS ENTRY ENGINE TORQUE
 AH-1G USA 9615247

SYM	DENSITY	ALT	GRWT	LONG. C.G.	ROTOR SPEED	SCAS	OAT	ARMAMENT
0	5000	7350	2010 (AFT)	324	ON	14	CLEAN	CONFIGURATION

NOTE 1. BROKEN LINE DENOTES MAX ANGLE OF SIDESLIP
 2. SOLID LINE DENOTES ANGLE OF SIDESLIP @ 1 SEC.
 3. CROSS HATCHING DENOTES SIDESLIP LIMIT

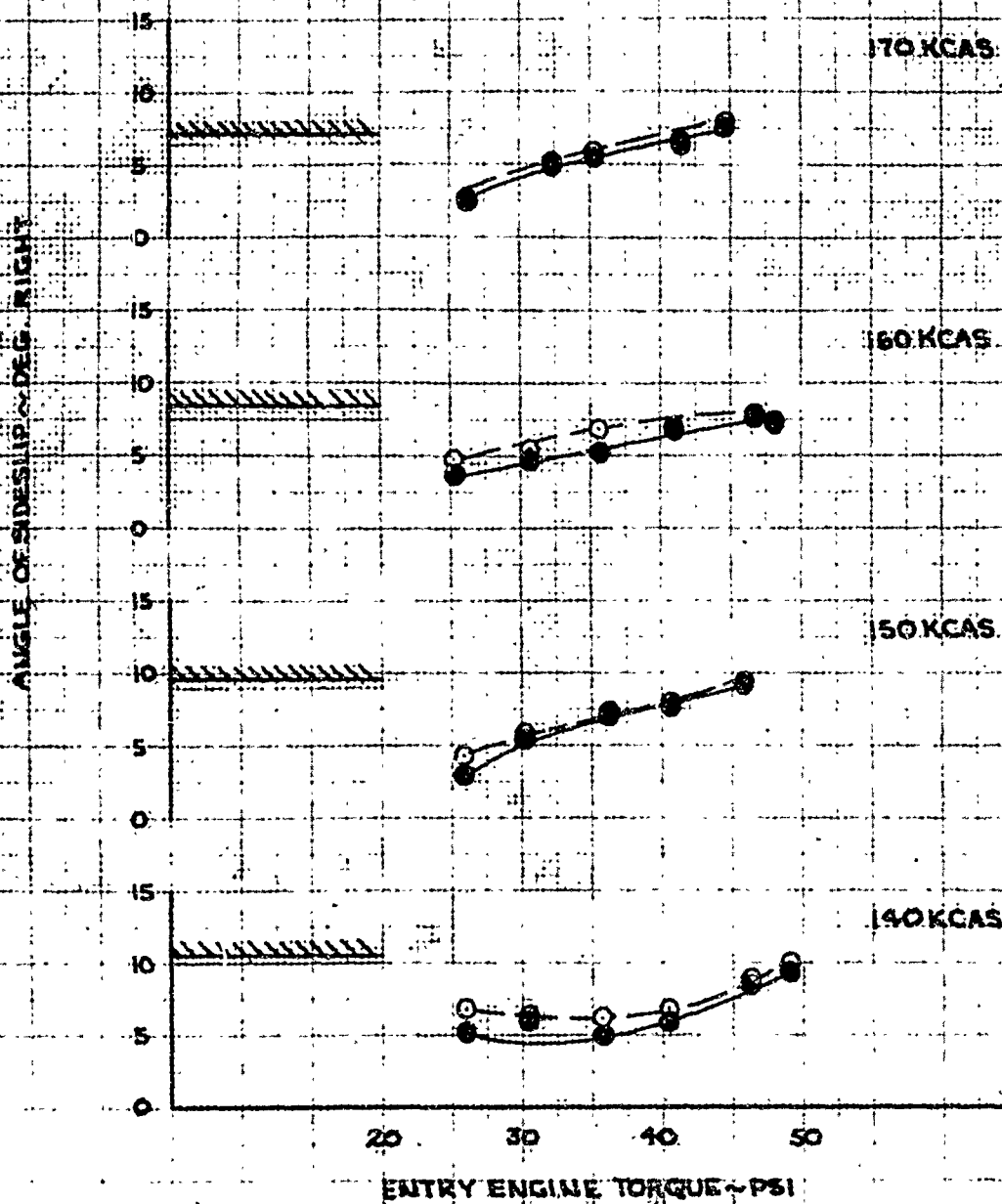


FIGURE 13
ANGLE OF SIDESLIP VS ENTRY ENGINE TORQUE
 AH-1G USA S/N 15247

SYM.	DENSITY	ALT.	GRWT	LONG. C.G.	ROTOR SPEED	SCAS	OAT	ARMAMENT
	H ₀ ~ FT		~ LB	~ IN.	~ RPM		~ °C	CONFIGURATION
Δ	5000		9200	192.7 (FWD)	324	ON	6	HVY. HOG

NOTE 1. BROKEN LINE DENOTES MAX ANGLE OF SIDESLIP
 2. SOLID LINE DENOTES ANGLE OF SIDESLIP @ 1 SEC.
 3. CROSS HATCHING DENOTES SIDESLIP LIMIT

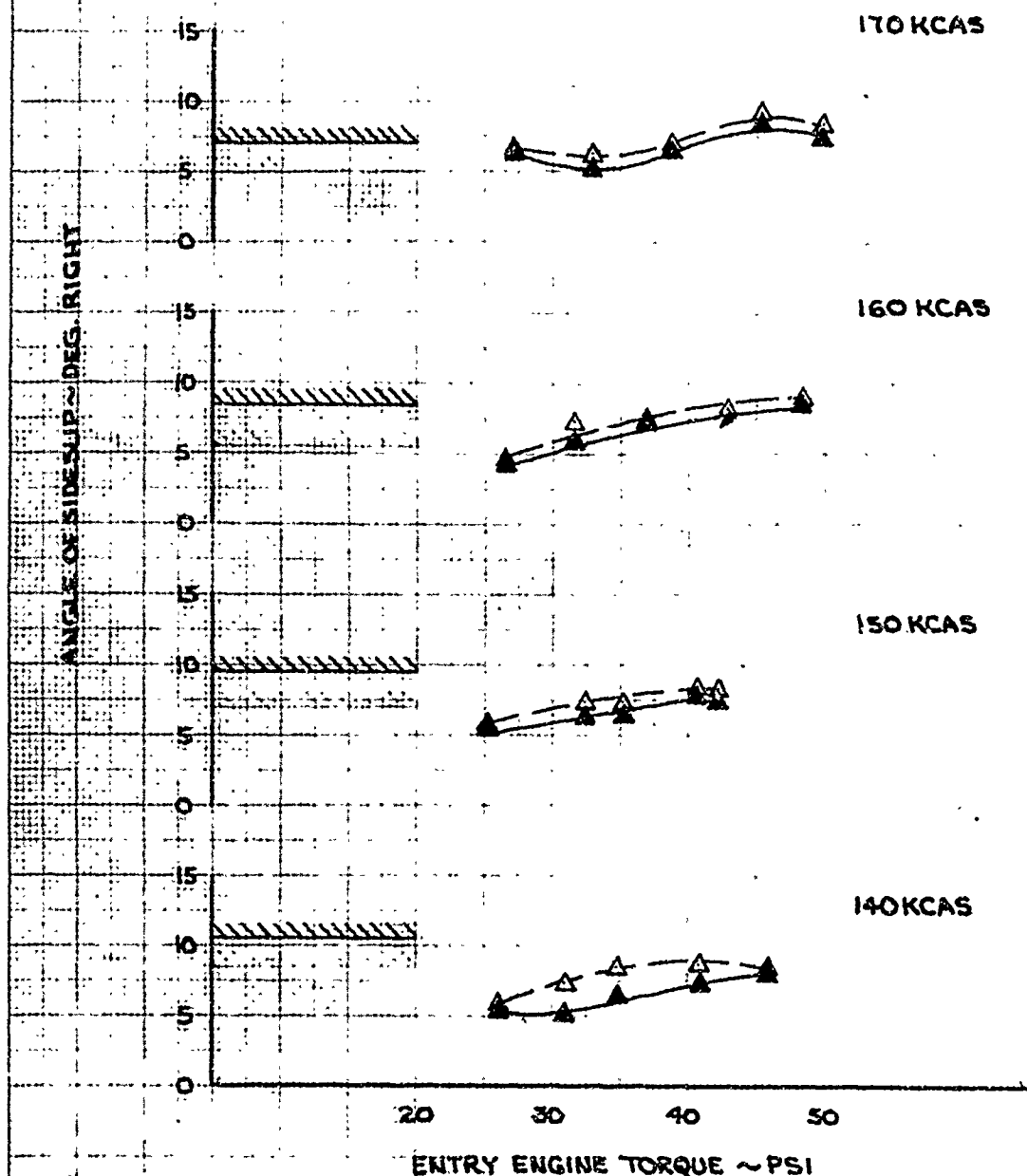


FIGURE 14
ANGLE OF SIDESLIP VS ENTRY ENGINE TORQUE
 AH-1G USA SUG 15247

SYM	DENSITY ALT. H _D ~ FT	GRWT ~ LB	LONG CG. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
□	5000	7200	201.0 (AFT)	324	ON	13	CLEAN
▽	5000	7200	201.0 (AFT)	324	OFF	13	CLEAN

NOTE: 1. BROKEN LINE DENOTES MAX. ANGLE OF SIDESLIP
 2. SOLID LINE DENOTES ANGLE OF SIDESLIP @ 1 SEC.
 3. CROSS HATCHING DENOTES SIDESLIP LIMIT

ANGLE OF SIDESLIP ~ DEG RIGHT

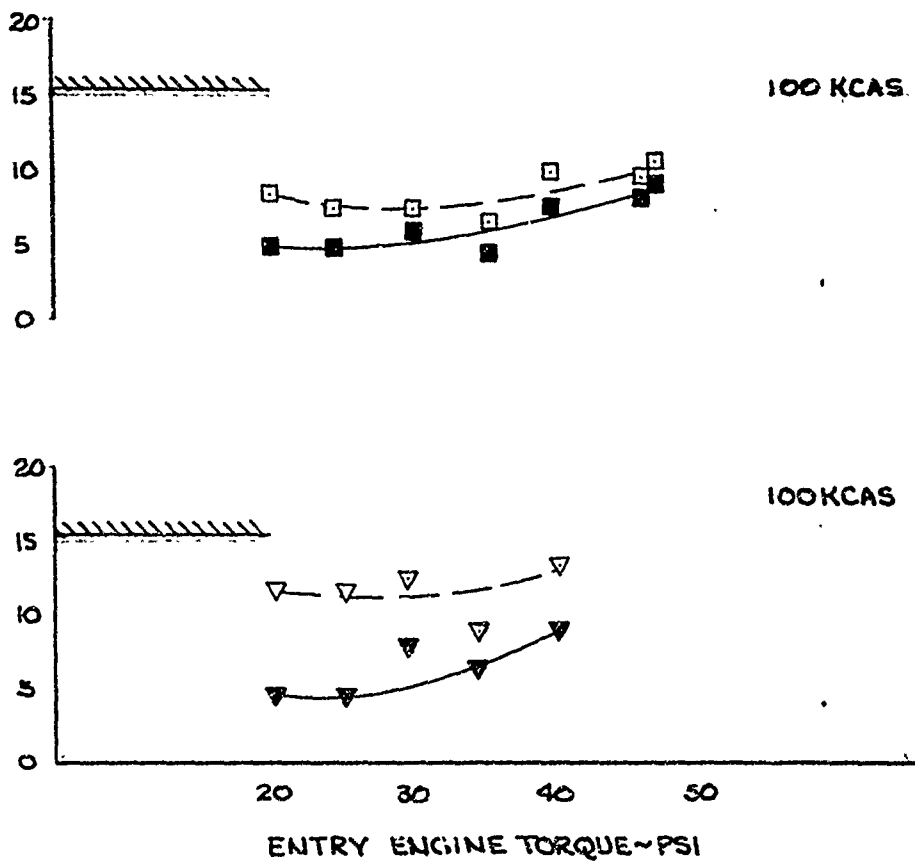


FIGURE 15
 ROTOR SPEED VS ENTRY ENGINE TORQUE

AH-1E USA S/N 615247

SYM DENSITY ALT. GRWT LONG.C.G. ROTOR SPEED SCAS OAT ARMAMENT
 H₀ ~ FT ~ LB. ~ IN. ~ RPM ~ °C CONFIGURATION
 ○ 5000 7350 201.0(AFT) 324 ON 14 CLEAN

NOTES: 1. SOLID LINE DENOTES ROTOR SPEED @ RECOVERY INITIATION
 2. BROKEN LINE DENOTES MINIMUM ROTOR SPEED
 3. POWER OFF TRANSIENT RPM LIMIT IS 250 RPM

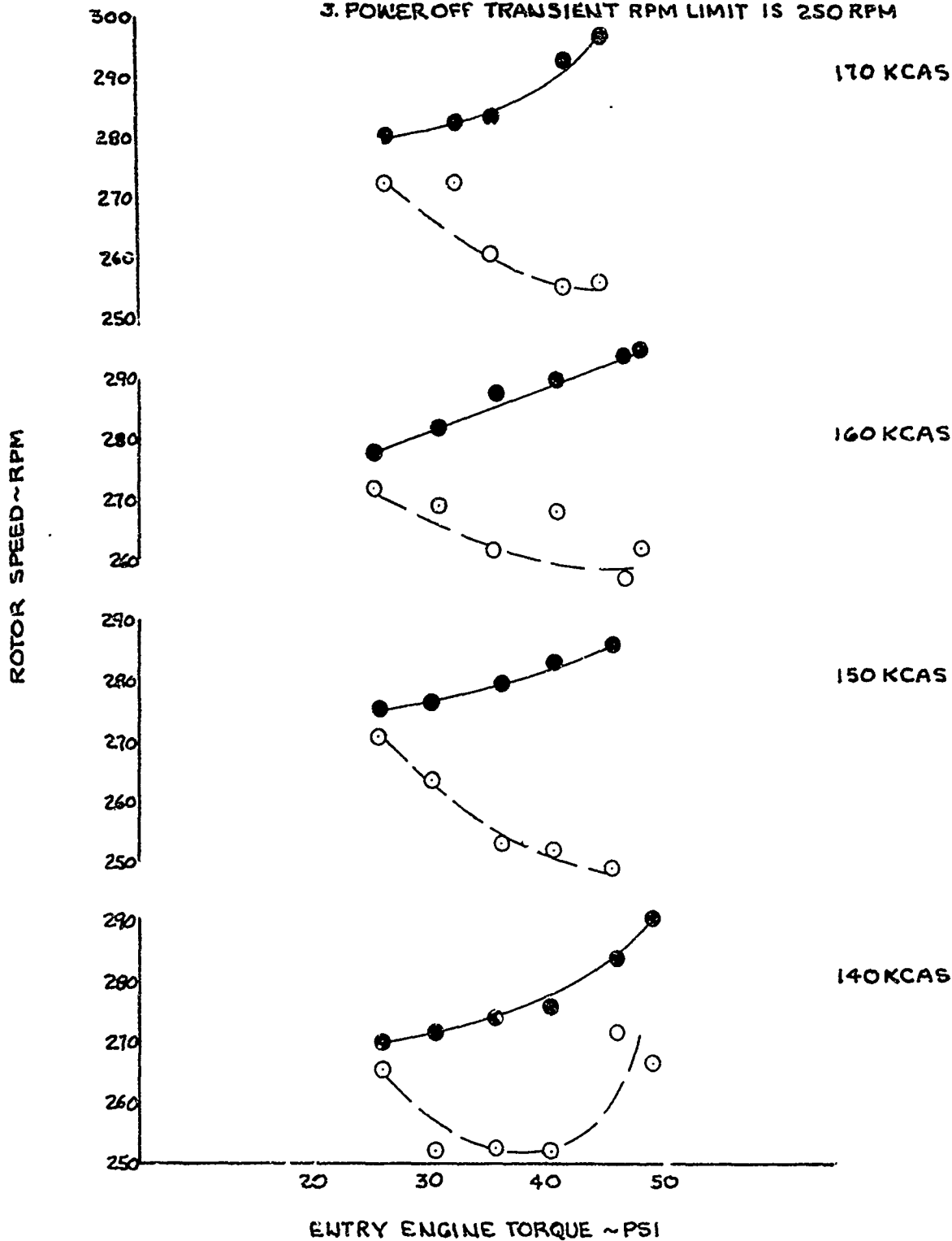


FIGURE 16
ROTOR SPEED VS ENTRY ENGINE TORQUE

AH-1G USA 9/N 615247

SYM	DENSITY ALT. H ₀ ~ FT.	GRWT ~ LB	LONG. C.G. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
Δ	5000	9200	192.7 (FWD)	324	ON	6	HVY. HOG

NOTES: 1. SOLID LINE DENOTES ROTOR SPEED @ RECOVERY INITIATION
2. BROKEN LINE DENOTES MINIMUM ROTOR SPEED
3. POWER OFF TRANSIENT RPM LIMIT IS 250 RPM

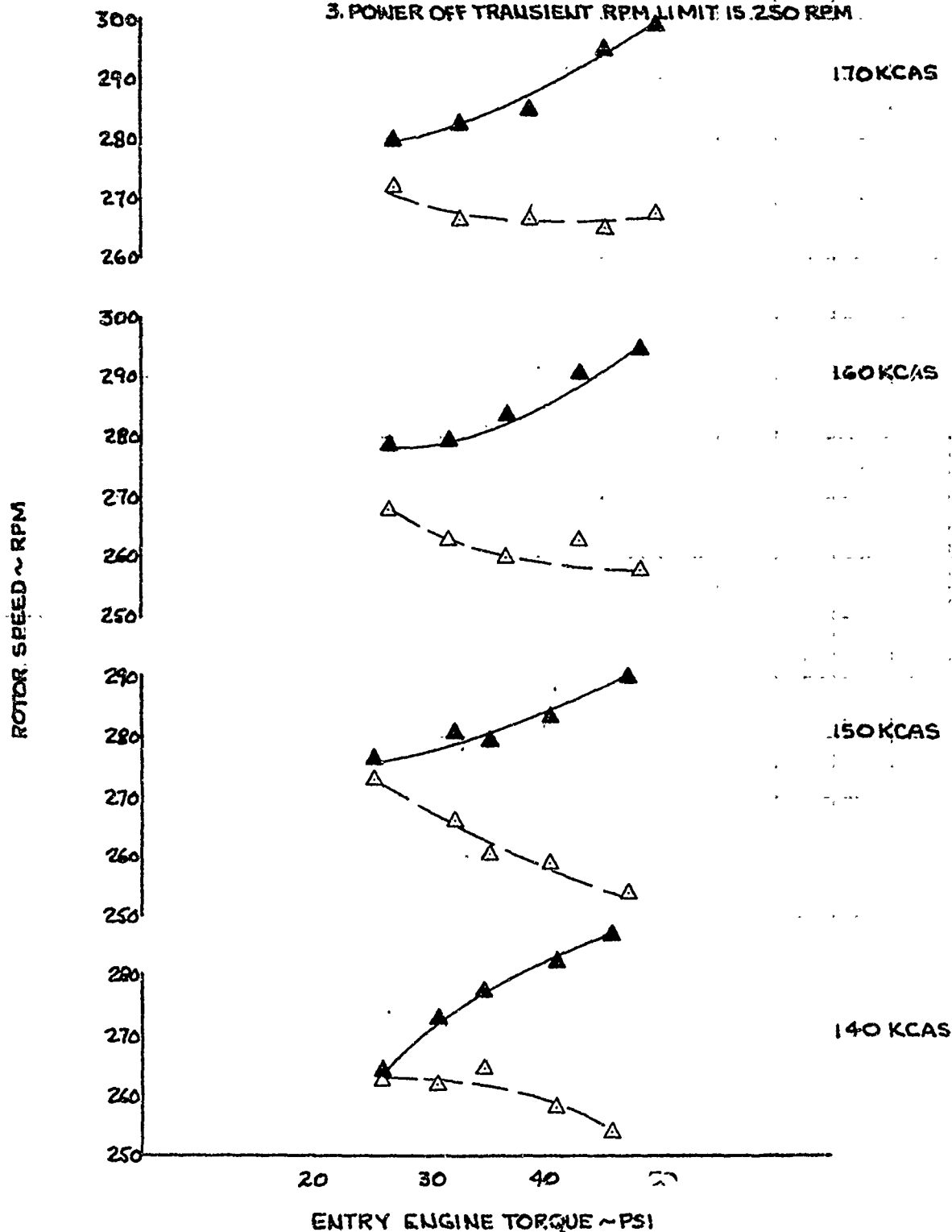


FIGURE 17
 ROTOR SPEED VS ENTRY ENGINE TORQUE

AH-1G USA S/N 615247

SYM.	DENSITY H ₀ ~ FT	ALT. ~ FT	GRWT. ~ LB.	LONG. C.G. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
□	5000		7200	201.0 (AFT)	324	ON	13	CLEAN
▽	5000		7200	201.0 (AFT)	324	OFF	13	CLEAN

NOTES: 1. SOLID LINE DENOTES ROTOR SPEED @ RECOVERY INITIATION
 2. BROKEN LINE DENOTES MINIMUM ROTOR SPEED
 3. POWER OFF TRANSIENT RPM LIMIT IS 250 RPM

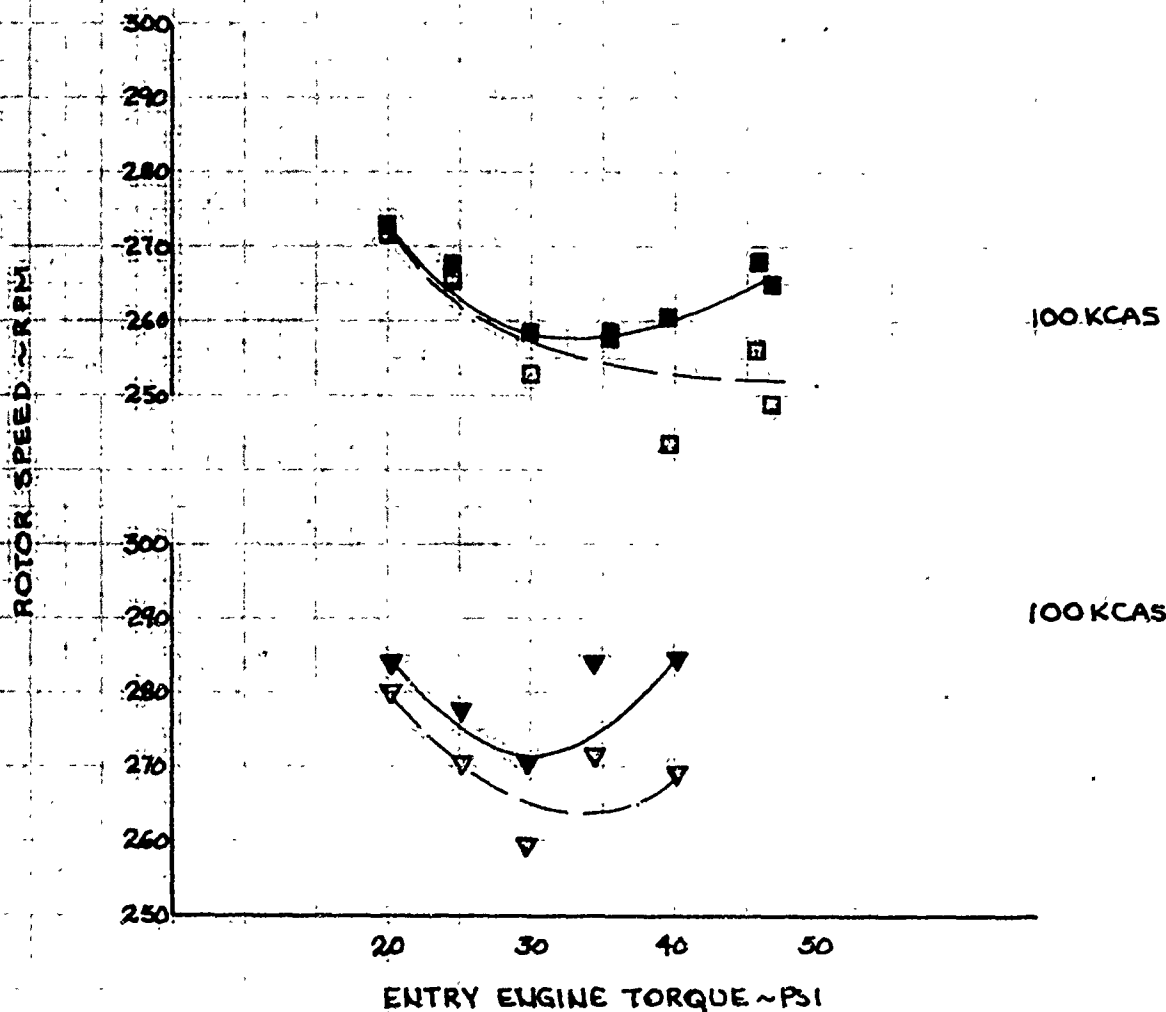


FIGURE 18
ROTOR SPEED DECAY RATE VS ENTRY ENGINE TORQUE
 AH-1G USA 34G15247

SYM	DENSITY ALT. H ₀ ~ FT	GRWT ~ LB	LONG. C.G. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
○	5000	7350	201.0 (AFT)	324	ON	14	CLEAN
△	5000	9200	192.7 (FWD)	324	ON	6	HVY. HOG
□	5000	7200	201.0 (AFT)	324	ON	13	CLEAN
▽	5000	7200	201.0 (AFT)	324	OFF	13	CLEAN

NOTE : DECAY RATES ARE AVERAGE

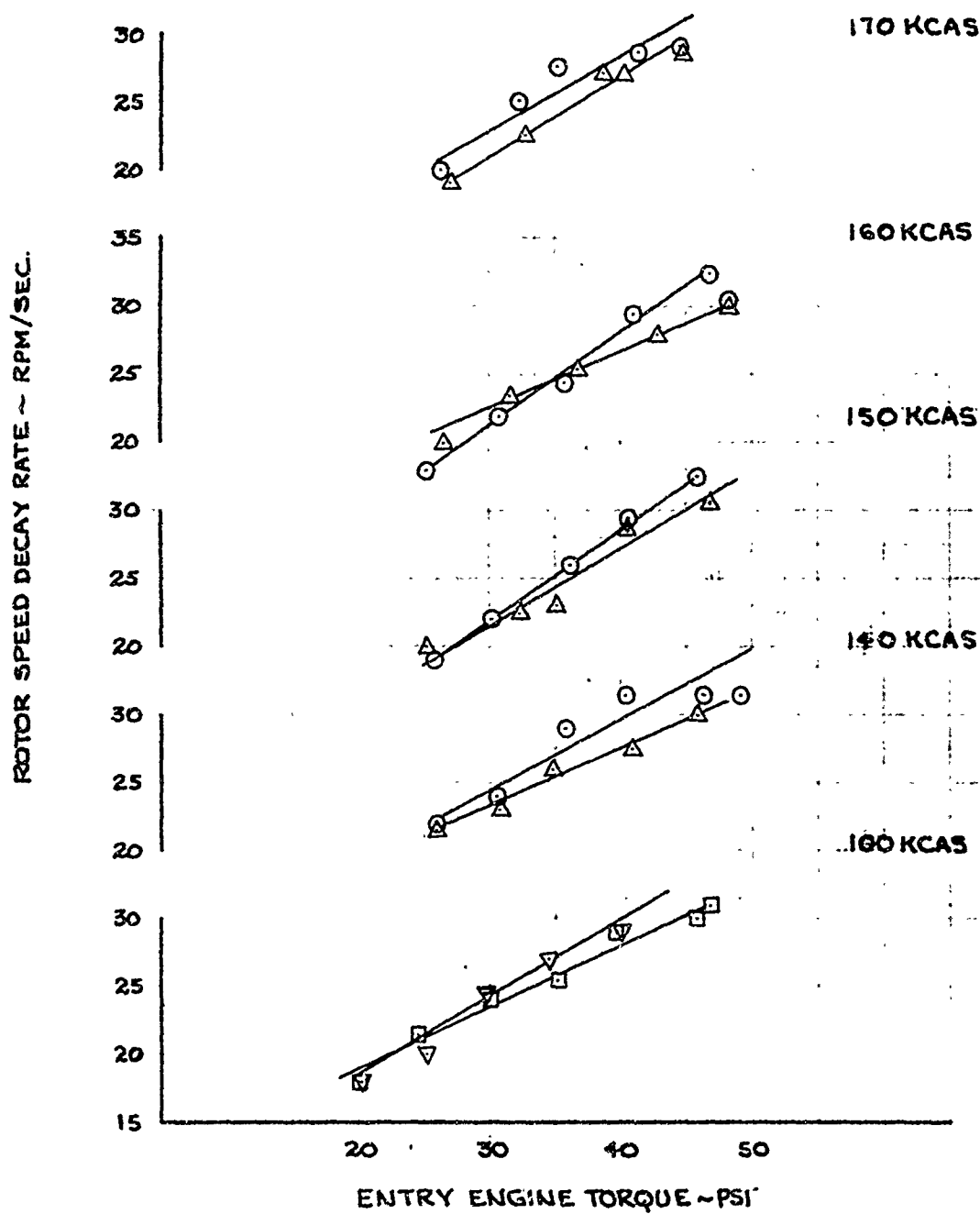


FIGURE 19
TIME TO 20° BANK ANGLE VS ENTRY ENGINE TORQUE
AH-1G USA 5/615247

SYM	DENSITY ALT. H _D ~ FT.	GRWT ~ LB	LONG. C.G. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
○	5000	7350	201.0 (AFT)	324	ON	14	CLEAN
△	5000	9200	192.7 (FW)	324	ON	6	HVY. HOG
□	5000	7200	201.0 (AFT)	324	ON	13	CLEAN
▽	5000	7200	201.0 (AFT)	324	OFF	13	CLEAN

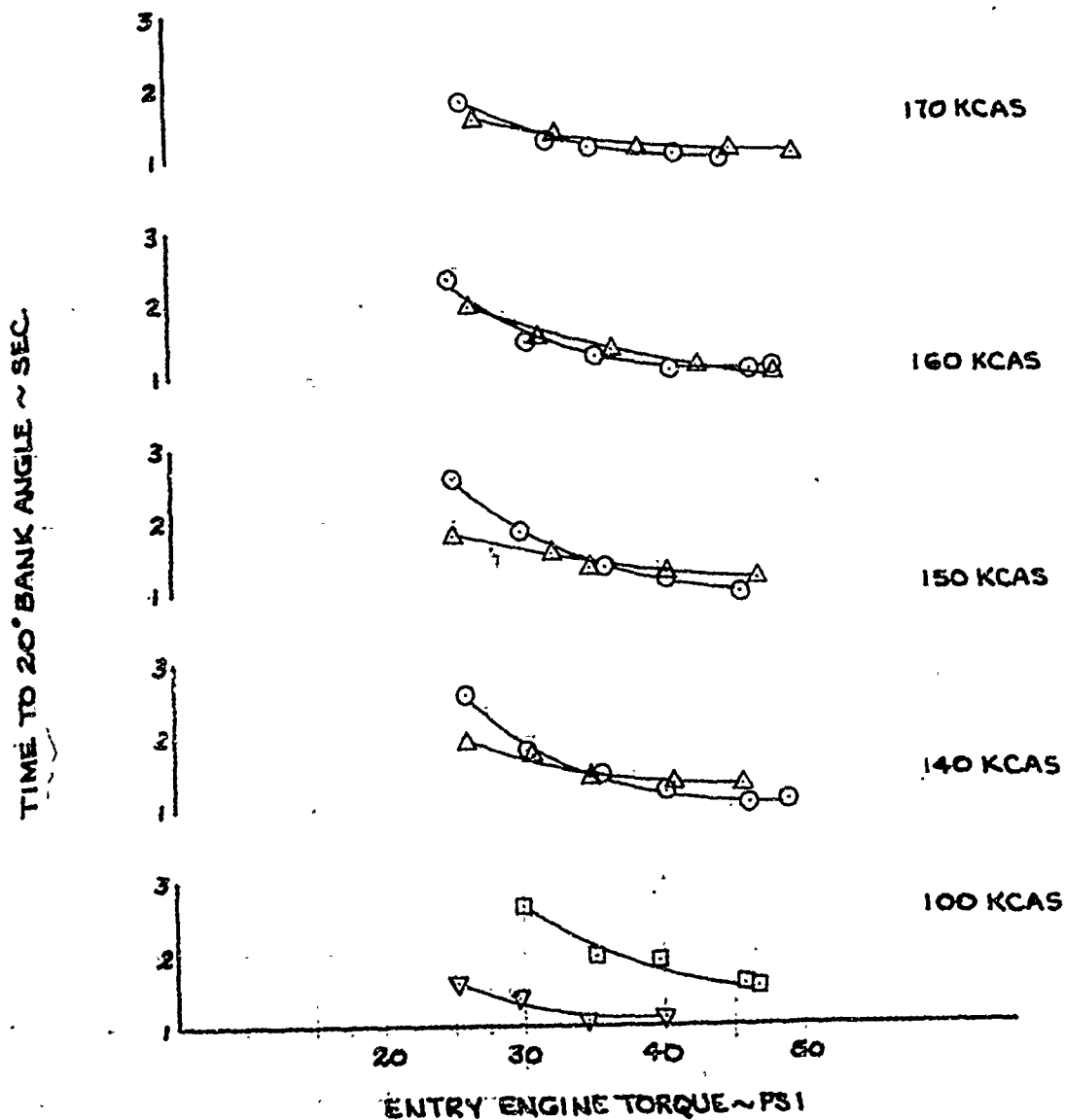


FIGURE 20
AIRCRAFT RESPONSE TO SIMULATED ENGINE FAILURE
AH-1G USA 5615247

ENGINE TORQUE AIRSPEED DENSITY ALT. GRWT. LONG CG. ROTOR SPEED SCAS OAT ARMAMENT
 ~ PSI ~ KCAS H₀ ~ FT. ~ LB ~ IN. ~ RPM OFF ~ ° CONFIGURATION
 40.1 100 5000 7200 201.0(AFT) 324 OFF 13 CLEAN

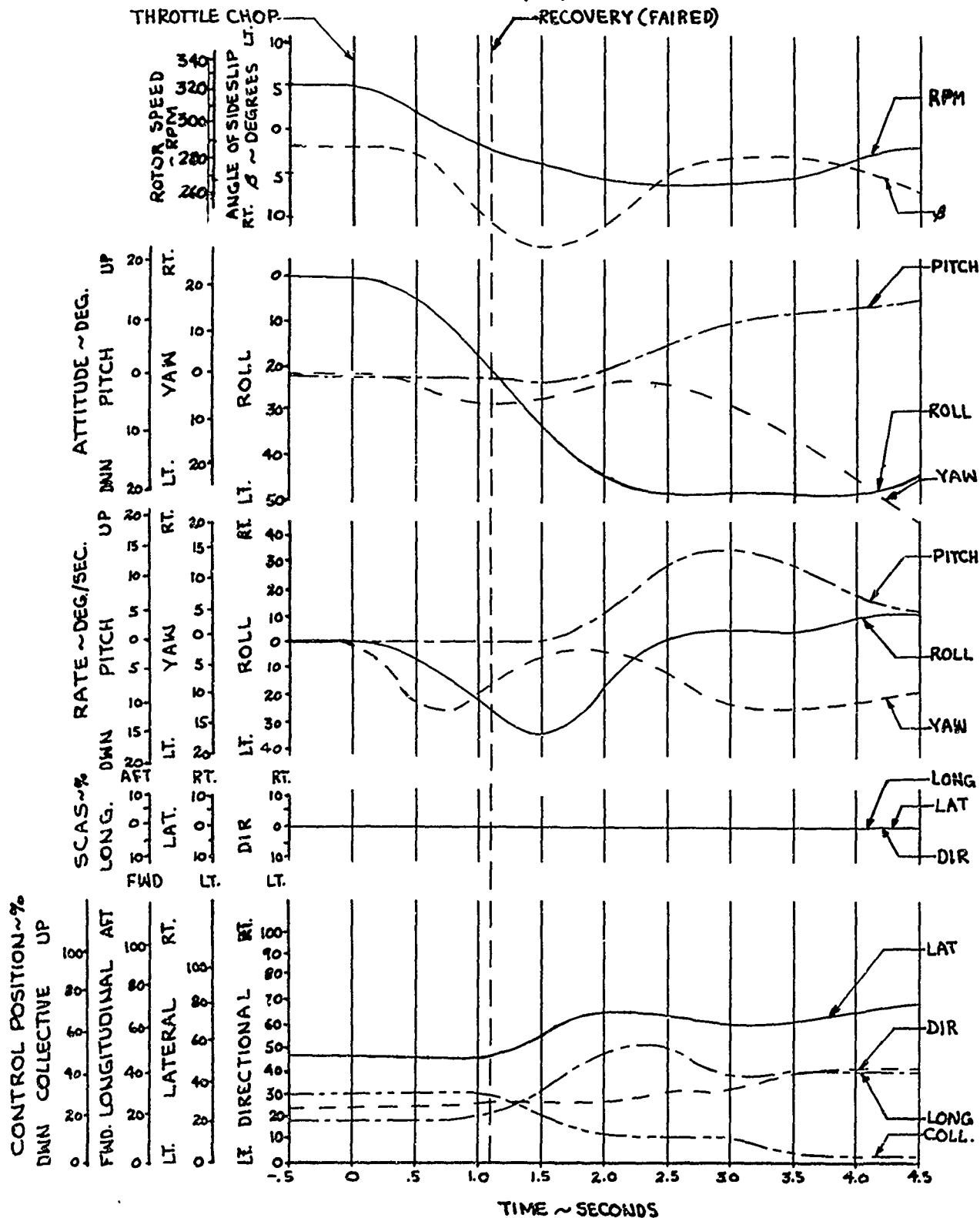


FIGURE 21
AIRCRAFT RESPONSE TO SIMULATED ENGINE FAILURE
AH-1G USAF 615247

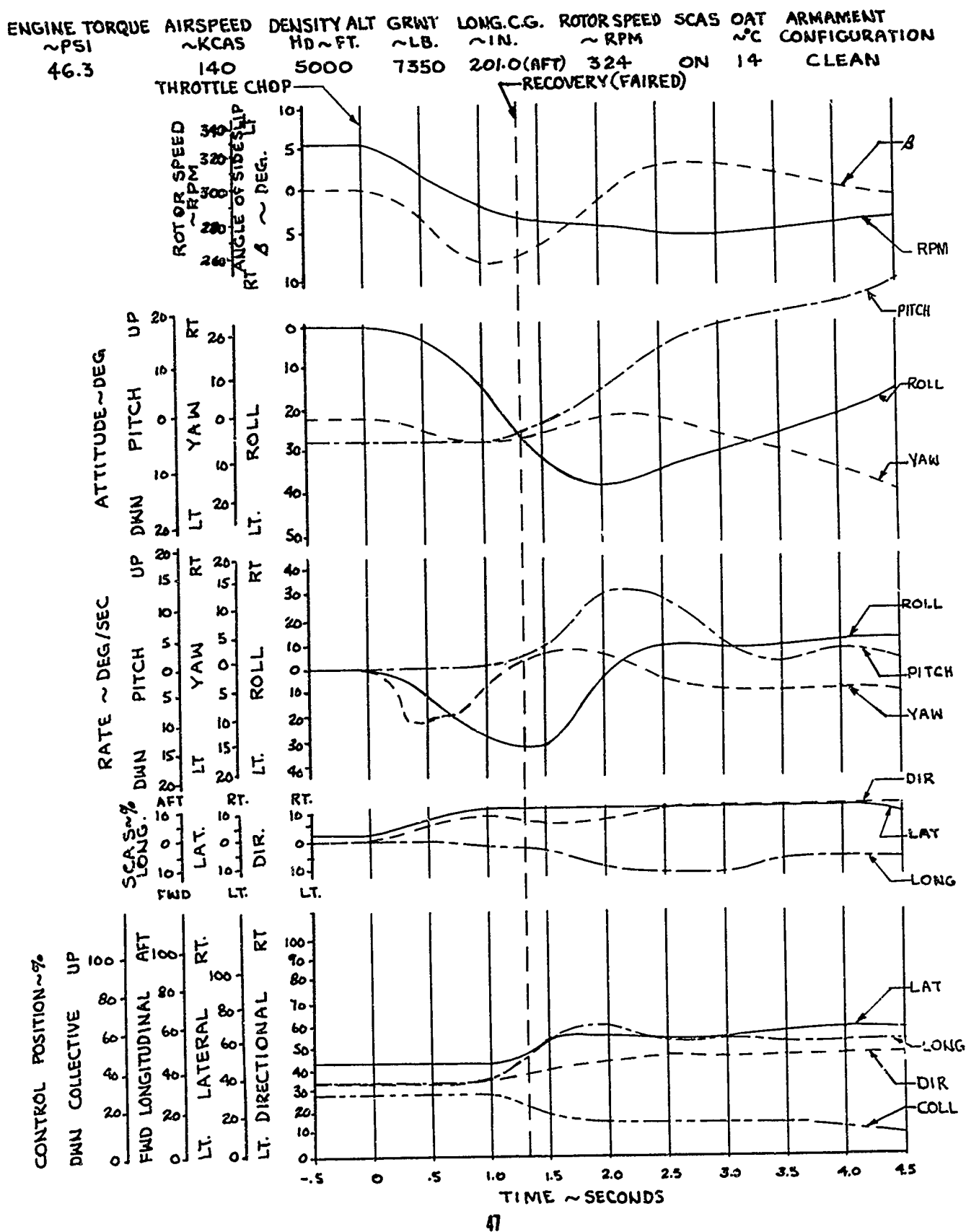


FIGURE 22
AIRCRAFT RESPONSE TO SIMULATED ENGINE FAILURE
AH-1G USA 5615247

ENGINE TORQUE ~ PSI	AIR SPEED ~ KCAS	DENSITY ALT. H ₀ ~ FT.	GRWT ~ LB	LONG. C.G. ~ IN.	ROTOR SPEED ~ RPM	SCAS ON	OAT ~ °C	ARMAMENT CONFIGURATION
40.6	150	5000	9200	192.7 (FWD)	324		6	HVY. HOG

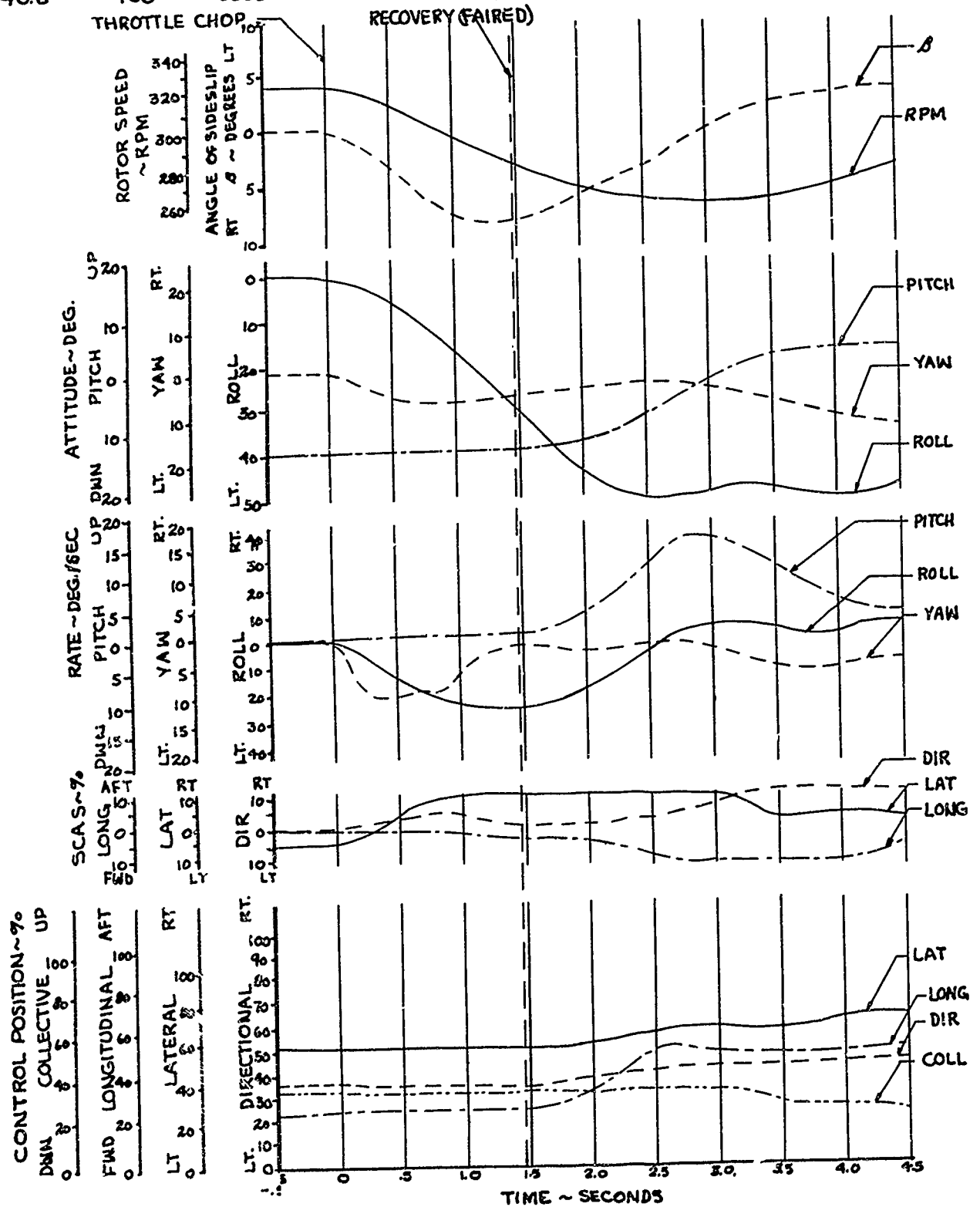


FIGURE 23
AIRCRAFT RESPONSE TO SIMULATED ENGINE FAILURE
AH-1G USA #615247

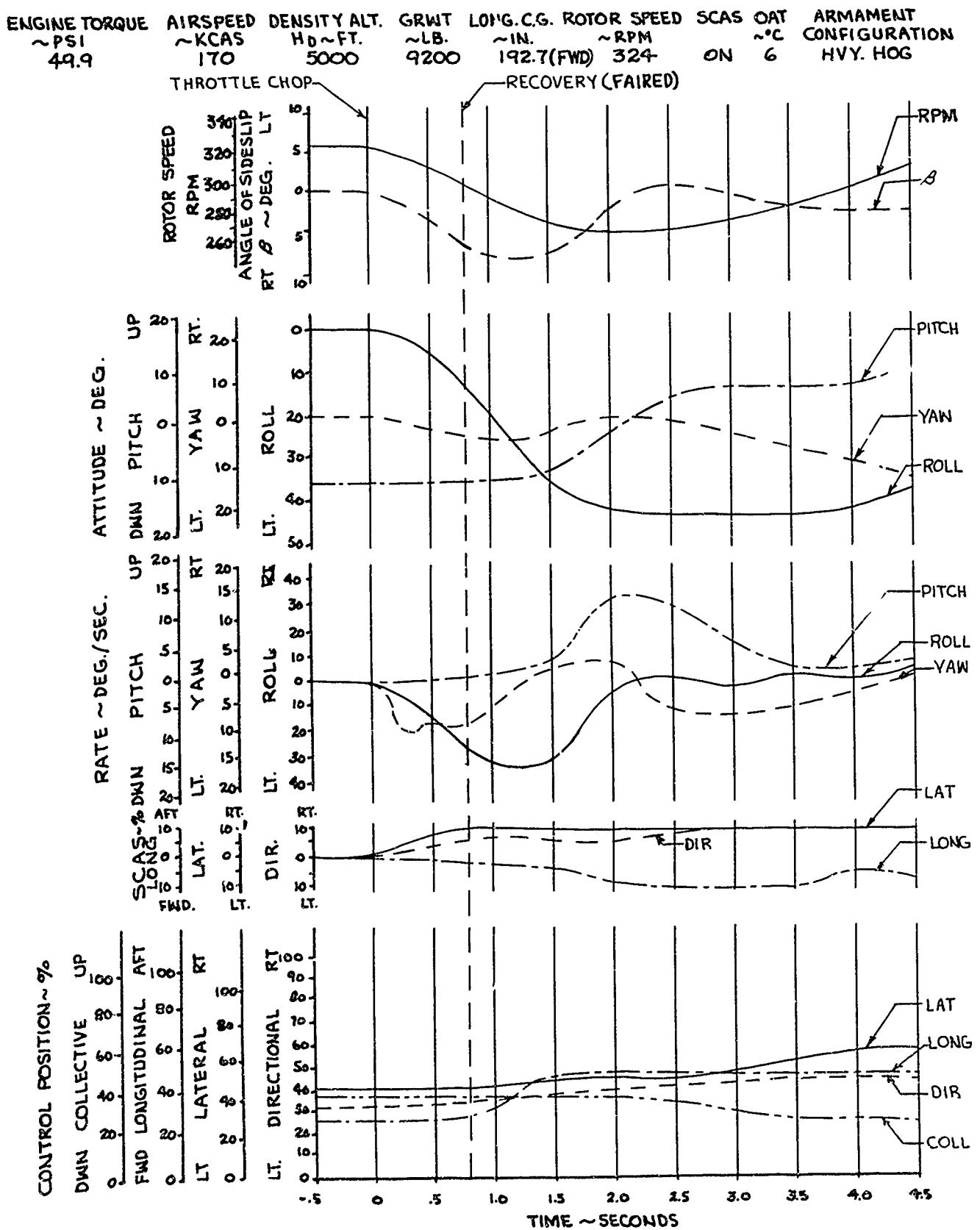


FIGURE 24
CONTROL DELAY TIME VS ENTRY ENGINE TORQUE
AH-1G USA SN 615247

SYM.	DENSITY ALT H _D ~ FT.	GRWT ~ LB	LONG. CG. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ~ °C	ARMAMENT CONFIGURATION
O	5000	7350	201.0 (AFT)	324	ON	14	CLEAN
Δ	5000	9200	192.7 (FWD)	324	ON	6	HVY. HOG

NOTE: RECOVERY INITIATED WHEN AIRCRAFT
RESPONSE REACHED THE TOLERANCE LIMIT

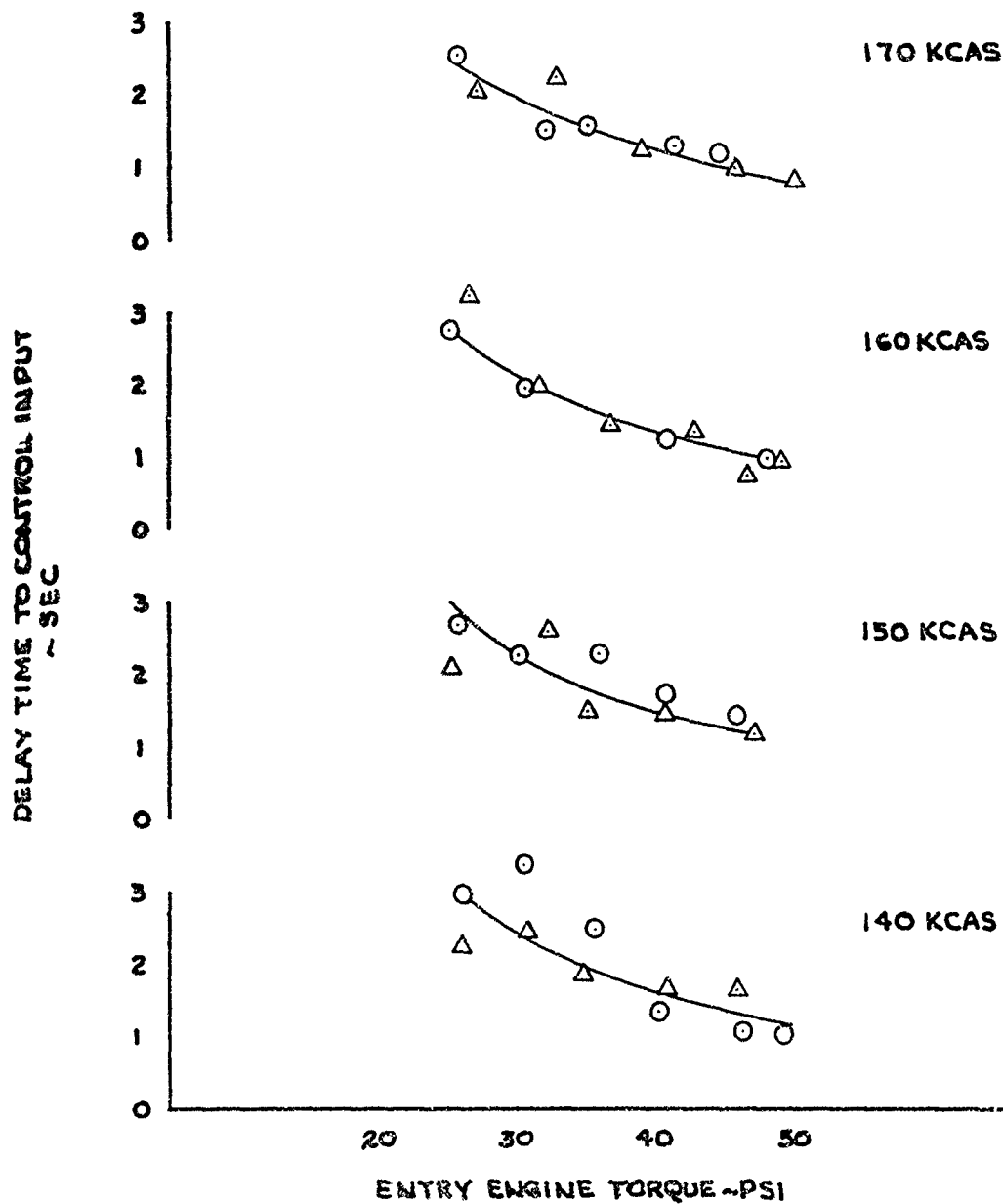


FIGURE 25
CONTROL DELAY TIME VS. ENTRY ENGINE TORQUE

AH-1G USA 5NG152A7

SYM	DENSITY ALT. H ₀ ~ FT	GRWT ~ LB	LONG. C.G. ~ IN.	ROTOR SPEED ~ RPM	SCAS	OAT ° C	ARMAMENT CONFIGURATION
□	5000	7200	201.0 (AFT)	324	ON	13	CLEAN
▽	5000	7200	201.0 (AFT)	324	OFF	13	CLEAN

NOTE: RECOVERY INITIATED WHEN AIRCRAFT
RESPONSE REACHED THE TOLERANCE LIMIT

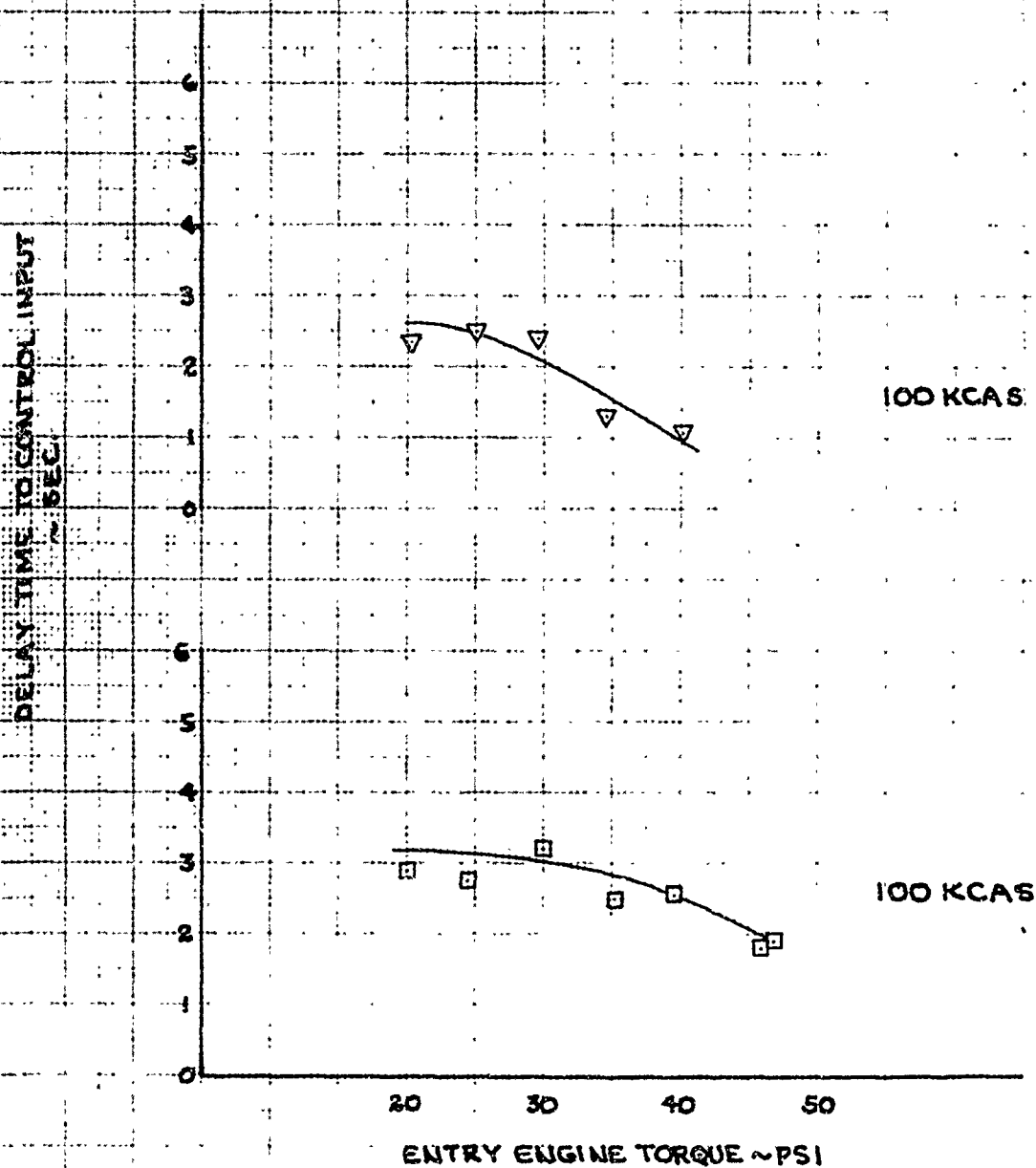


FIGURE 26
ENTRY ENGINE TORQUE VS ENTRY AIRSPEED
 AH-1G USA W/G 15247

NOTE: CURVES DENOTE MAXIMUM CONTROL INPUT DELAY TIMES
 FOLLOWING SIMULATED ENGINE FAILURE FOR VARIOUS
 AIRCRAFT RESPONSE TOLERANCES

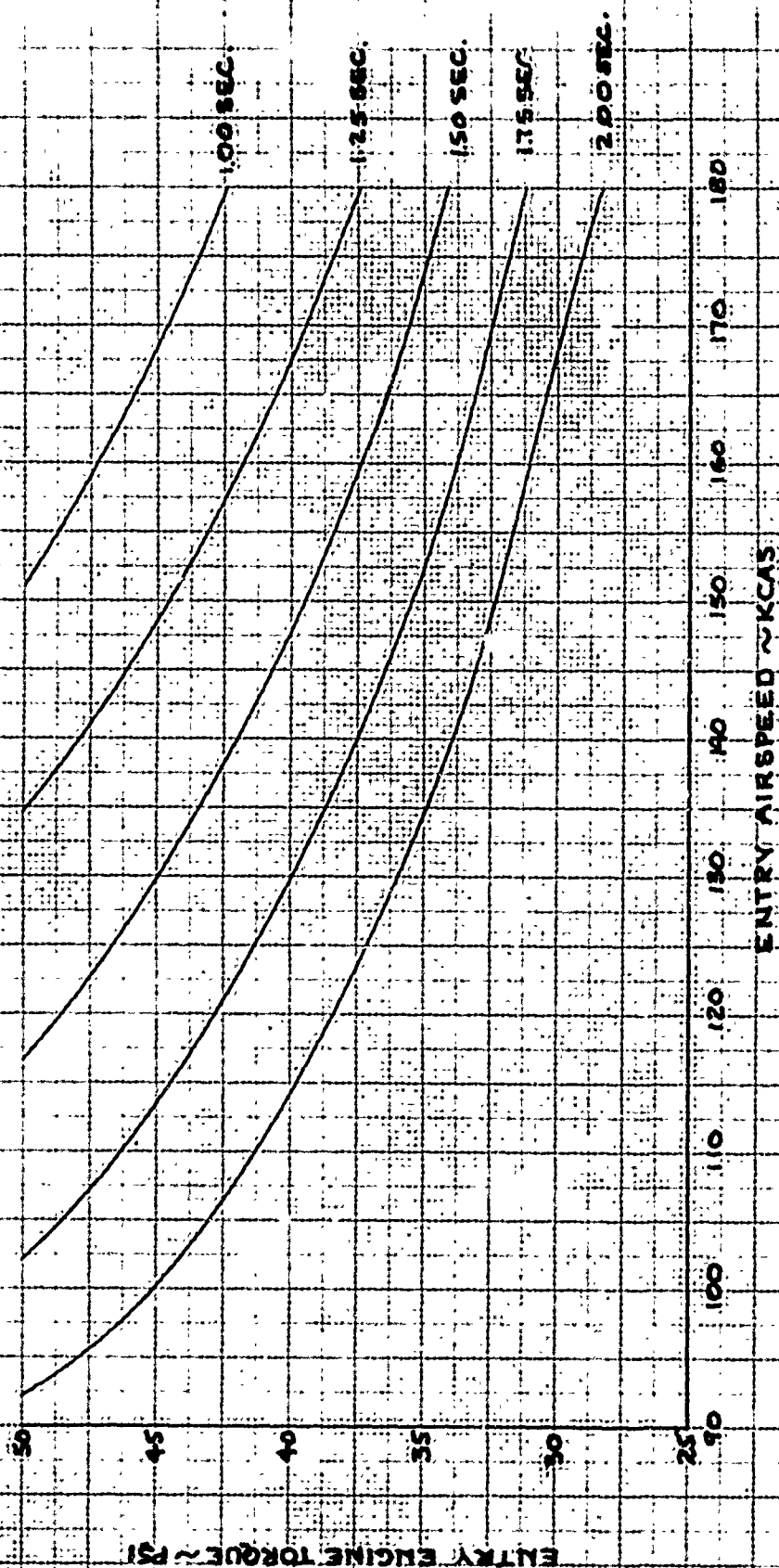


FIGURE 27
SUDDEN ENGINE FAILURE ENVELOPE
AH-1G

CAUTION: SUDDEN ENGINE FAILURE WILL CAUSE A RAPID ROLL TO THE LEFT AND RAPID ROTOR RPM DECAY IF PROPER RECOVERY INPUTS ARE NOT MADE WITHIN 1 SECOND WHEN OPERATING AT HIGH ENGINE TORQUE AND HIGH AIRSPEED

