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HEIGHT-VELOCITY EVALUATION
CH-47C HELICOPTER WITH T55-L-11A ENGINES

FINAL REPORT

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

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ABSTRACT

The CH-47C height-velocity flight test program was conducted at Edwards Air Force Base and Shafter, California, and Tonopah Test Range, Nevada, between 29 September 1971 and 9 March 1972. Engineering flight tests were conducted to develop realistic single-engine height-velocity diagrams for the CH-47C helicopter with T55-L-11A engines. During these tests, no deficiencies were identified, but one shortcoming was identified: the excessive pilot compensation required to control pitch attitude following a simulated single-engine failure from an out-of-ground-effect hover. The height-velocity diagrams developed are suitable for inclusion in the operator's manual when accompanied by the flight conditions and a discussion of the pilot technique. Entry characteristics of the helicopter following engine failure are satisfactory. Power settling may occur following an engine failure from an out-of-ground-effect hover unless the helicopter is pitched immediately to an accelerating attitude before the thrust control rod is lowered. The takeoff procedures depicted in the operator's manual and the US Army Aviation School CH-47 standardization guide are safe in the event of single-engine failure. However, hard landings may result when a power failure occurs during a steep approach at or above a 40,800-pound gross weight or during a normal approach at a 46,000-pound gross weight. Increases in gross weight and density altitude degraded height-velocity performance. Efforts to generalize height-velocity performance data using analytical procedures and referred-gross-weight methods were unsuccessful. Height-velocity performance was apparently unaffected by the center-of-gravity location or which engine was failed. Further testing at high outside air temperatures would be required to completely define the single-engine height-velocity performance of the CH-47C helicopter equipped with T55-L-11A engines.

TABLE OF CONTENTS

Page

APPENDIXES

INTRODUCTION

BACKGROUND

1. Single-engine height-velocity (H-V) testing has not previously been conducted with the CH-47C helicopter. The operator's manual for this aircraft (ref 1, app A) does not contain an H-V diagram. Single-engine H-V testing of the CH-47B has been conducted at gross weights up to 40,000 pounds (refs 2 and 3); however, these test results are not directly applicable to the CH-47C. The US Army Aviation Systems Command (AVSCOM) directed the US Army Aviation Systems Test Activity (USAASTA) to conduct height-velocity tests on the CH-47C (ref 4). The CH-47C height-velocity test plan (ref 5) was prepared in accordance with the test directive.

TEST OBJECTIVES

2. The objectives of the CH-47C H-V test were as follows:

a. To develop operational CH-47C H-V diagrams for incorporation in the operator's manual.

b. To determine compliance with the military specification, MIL-H-8501A (ref 6, app A), and the detail specification (ref 7).

DESCRIPTION

3. The CH-47C helicopter is manufactured by the Vertol Division of The Boeing Company (Boeing-Vertol). It is a twin-turbine, tandem-rotor helicopter designed to provide air transportation for cargo, troops, and weapons. The helicopter is intended for use during visual or instrument flight conditions. The test helicopter was powered by two T55-L-11A Lycoming engines. A more complete description of the CH-47C is presented in the operator's manual (ref 1, app A) and in appendix B.

SCOPE OF TEST

4. Height-velocity tests were conducted with the CH-47C helicopter from 29 September to 9 March 1972 at Edwards Air Force Base (2302-foot elevation) and Shafter, California (420-foot elevation), and Tonopah Test Range, Nevada (5540-foot elevation). During the test program, 47 flights were conducted for a total of 48 hours, of which 34 hours of productive testing were accomplished. Testing at a safe altitude above ground level (AGL) was accomplished at gross weights from 29,400 to 46,000 pounds, density altitudes from 2000 to 6000 feet,

and center-of-gravity (cg) locations from fuselage station (FS) 319.5 (forward) to FS 334.5 (aft). Height-velocity tests were accomplished to a touchdown at the conditions shown in table 1. The scope of this evaluation was limited to single-engine failures. The tests were conducted to produce data which were realistic with respect to operational conditions and do not show the maximum capability of the aircraft.

Table 1. Height-Velocity Touchdown Test Conditions.

Average Gross Weight (lb)	Average Density Altitude (ft)	Average Outside Air Temperature (°C)	Average Center-of-Gravity Location (in.)	Entry Flight Condition
40,910	4500	-2.5	FS 325.8 (mid)	Level flight
44,110	4500	-2.0	FS 326.2 (mid)	Level flight
46,100	4030	-9.0	FS 327.3 (mid)	Level flight
40,870	650	-1.0	FS 325.8 (mid)	Level flight
44,080	-200	-5.5	FS 326.2 (mid)	Level flight
46,050	900	4.5	FS 327.3 (mid)	Level flight
40,870	650	-1.0	FS 325.8 (mid)	Takeoff
46,030	1150	3.5	FS 327.3 (mid)	Takeoff
41, 40	480	-0.5	FS 325.8 (mid)	Approach
46,030	1150	3.5	FS 327.3 (mid)	Approach

5. All of the CH-47C H-V tests were conducted without external loads. Ballasting was accomplished by use of internal water tanks. The cargo hook and the lower rescue door were removed to facilitate emergency water jettison. The pitch stability augmentation (PSA) system was placed in the automatic-synchronization mode because that is the normal operational mode.

6. Maximum-rated power checks (topping) were accomplished on the installed engines in accordance with the current maintenance procedures stated in the CH-47C organizational maintenance manual (ref 8, app A). The allowable range for the indicated torque is ±1 percent. The engines were adjusted so that power available was in the lower half of this range. This was to ensure that the test aircraft did not have more power available than representative operational aircraft.

7. The flight restrictions and limitations contained in the safety-of-flight release (ref 9, app A) were observed. All H-V touchdowns were accomplished on a paved surface.

METHODS OF TEST

8. The procedure used to simulate a sudden engine failure was to stabilize the helicopter at the desired conditions of airspeed and height AGL, and then to place one engine condition lever in the ground position. This was accomplished following a countdown and the simulated failure was not a surprise to the pilot. The delay time between power reduction and control movement was started when the condition lever reached the ground position. Except as noted in appendix C, conventional H-V test techniques and data analysis procedures were used as discussed in reference 10, appendix A. A Handling Qualities Rating Scale (HQRS) was used to augment qualitative comments (app D).

9. To provide a realistic H-V diagram of maximum benefit to the operational aviator, the pilot technique was critiqued by two aviators with extensive CH-47 experience in the field and by a recent graduate of the CH-47 transition course at the US Army Aviation School, Fort Rucker, Alabama. These aviators were placed in several test conditions at altitude and their reactions and techniques were recorded. The test technique developed by the test team was then demonstrated to these aviators and a final technique was developed based on their comments.

10. The tests were conducted under nonturbulent atmospheric conditions to produce accurate, repeatable data. All touchdown tests were conducted in wind velocities of 5 knots or less. The test CH-47C helicopter (serial number 68-15859) was equipped with sensitive, calibrated instrumentation. A detailed list of the test instrumentation is presented in appendix E.

CHRONOLOGY

11. The chronology of the test program is listed below. The delay in the start of H-V testing was due to other CH-47C testing.

Height-velocity test request received	3	November	1969
Height-velocity test flying commenced	29	September	1971
Height-velocity test flying completed	9	March	1972

RESULTS AND DICUSSION

GENERAL

12. Engineering flight tests were conducted to develop realistic single-engine height-velocity diagrams for the CH-47C helicopter with T55-L-11A engines. During these tests, no deficiencies were identified, but one shortcoming was identified: the excessive pilot compensation required to control pitch attitude following a simulated single-engine failure from an out-of-ground-effect hover. The height-velocity diagrams developed are suitable for inclusion in the operator's manual when accompanied by the flight conditions and a discussion of the pilot technique. Entry characteristics of the helicopter following engine failure are satisfactory. Power settling may occur following an engine failure from an out-of-ground-effect hover unless the helicopter is pitched immediately to an accelerating attitude before the thrust control rod is lowered. The takeoff procedures depicted in the operator's manual and the US Army Aviation School CH-47 standardization guide are safe in the event of single-engine failure. However, hard landings may result when a power failure occurs during a steep approach at or above a 40,800-pound gross weight or during a normal approach at a 46,000-pound gross weight. Increases in gross weight and density altitude degraded height-velocity performance. Efforts to generalize height-velocity performance data using analytical procedures and referred-gross-weight methods were unsuccessful. The center-of-gravity location and the particular engine which was selected to remain operational had no apparent effect on height-velocity performance. Further testing at high outside air temperatures would be required to completely define the single-engine height-velocity performance of the CH-47C helicopter equipped with T55-L-11A engines.

ENTRY CHARACTERISTICS

13. Sudden single-engine failures were simulated by first stabilizing the aircraft at the desired conditions of airspeed and height above the ground, and then placing one engine condition lever in the ground position. Simulated single-engine failures were conducted at gross weights from 32,000 to 46,000 pounds and airspeeds from hover to 152 knots calibrated airspeed (KCAS). Following a simulated single-engine failure, with the PSA system in the automatic-synchronization mode, a small nose-up pitch change occurred and the aircraft stabilized at the new pitch attitude. The helicopter slowly rolled to the left at a rate that was easily controlled by the pilot. At airspeeds above 130 KCAS, the roll rate was slightly higher. During the stability and control tests (ref 11, app A), a slow, divergent nose-up pitch change resulted from a simulated single-engine failure with the PSA system OFF, but the pitch-up was easily controlled by the pilot. Testing showed that for similar aircraft conditions and engine topping settings, single-engine H-V performance was similar for each engine. During the actual H-V diagram development, the engine failure was simulated by reducing power on the left engine.

14. Following a single-engine failure, the operating engine increased power until reaching maximum power available for the engine beep trim setting. The minimum transient rotor speeds during the control-fixed period following the simulated failure are presented in figure 1, appendix F. The data show that for the conditions tested, the minimum transient rotor speed did not reach generator cutoff rotor speed of 204 ± 4 rpm. During all the level flight entry tests, the rotor speed decay rate increased with gross weight and decreased with forward speed. The rotor speed stabilized above 210 rpm following all simulated single-engine failures in level flight without moving the thrust control rod or increasing engine beep trim. Within the scope of this test, the CH-47C entry characteristics following a simulated single-engine failure are satisfactory.

15. Entry characteristics following simulated dual-engine failures were evaluated during the CH-47C stability and control tests (ref 11, app A) using essentially the same methods as for single-engine failures, except that both engine condition levers were placed in the ground position. The flight controls were held fixed as long as practical after the simulated failure. These tests were conducted at gross weights from 33,000 to 45,000 pounds and airspeed from 78 to 148 KCAS. Results of the tests indicate that at airspeeds below 100 knots, there was a slight nose-up pitch which was easily corrected. There was no apparent roll attitude change. Response of the helicopter following simulated dual-engine failures at airspeeds greater than 100 KCAS was more severe than under simulated single-engine failures. At airspeeds of 100 KCAS or less, the response was similar to the single-engine failure response. The nose-up pitch change following a dual-engine failure was adequately corrected by the PSA system. With the PSA system OFF, a correction of the divergent nose-up pitching required a slightly faster pilot reaction than was required with single-engine failure, but presented no aircraft control problem. Lateral and directional oscillations were apparent following failures at airspeeds in excess of 140 KCAS, but did not limit control of the aircraft. The noise change associated with the rapid rotor speed decay provided the pilot with an unmistakable cue to an engine failure. Time delays from engine failure to thrust control rod movement were slightly in excess of 1 second and produced a minimum transient rotor speed of approximately 190 rpm. At the minimum transient rotor speed there was no apparent degradation in controllability. These response characteristics and delay times between dual-engine failure and thrust control rod movement, evaluated during the previous stability and control testing, met the requirements of the detail specification and are satisfactory.

PILOT TECHNIQUE FOLLOWING ENTRY

16. There are many techniques which could be used to transition from full-power flight to steady-state autorotational flight. The variables were too numerous for an exhaustive evaluation during this program to determine the technique which provides the maximum capability of the aircraft for the entire envelope. The pilot technique determination was therefore limited to the range of conditions which were within the capabilities (training and tolerance) of operational pilots.

17. In discussion with instructor pilots at the US Army Aviation School and with two experienced CH-47 pilots at USAASTA, the consensus was that nose-down pitch attitudes beyond 20 degrees or rates in excess of 20 degrees per second were extreme and could not be consistently expected from operational pilots. To determine the reaction or corrective action which could be expected from operational pilots and therefore would be best for the conduct of this test, simulated single-engine failures were conducted at a safe altitude above the ground at various gross weights, airspeeds, and density altitudes. At gross weights above 40,000 pounds, at airspeeds from hover to 42 KCAS, and using a similar pitch rate, increasing nose-down pitch attitudes decreased the height loss to reach a specific airspeed. Height loss also decreased as higher pitch rates were used to reach a given pitch attitude. Time histories of pitch attitudes and rates used following simulated single-engine failures at different entry airspeeds are presented in figure 2, appendix F. Pitch attitudes and maximum pitch rates used during the touchdown tests are presented in figure 3. The recommended pitch attitudes and rates are presented in table 2. At airspeeds above 80 KCAS, manipulation of the cyclic and thrust control rods to slow the helicopter to a safe touchdown speed while maintaining approximately 235 rpm was necessary.

Table 2. Recommended Pitch Attitudes and Rates.

Calibrated Airspeed at Engine Failure (kt)	Stabilized Pitch Attitude (deg nose down)	Maximum Pitch Rate (deg/sec nose down)
Hover	17	16
42	12	9
58	9	7

18. The best pilot cue to engine failure is the sound change associated with decreasing rotor speed. Less obvious cues are the torque split and engine compressor speed (N_1) decay as observed on the appropriate instruments. The time required for pilot recognition and reaction was estimated at 2 seconds and the audio cue of rotor speed decay is consistent with that estimate. All tests, except during takeoff and approach, incorporated a 2-second delay from the time the engine condition lever was moved to the ground position until movement of the flight controls. During takeoff and approach tests, a zero time delay was used to more closely simulate operational flying where aircraft operation is more closely monitored.

19. The normal engine trim control switch (beep) was used to gain maximum available power on the operating engine following the simulated failure. It was determined prior to the touchdown tests that this procedure would be unsafe for testing since it placed the power turbine actuator out of the rotor speed governing range, which could easily result in rotor overspeed following the landing. Therefore, the test technique was to adjust the thrust control rod and not the engine beep

trim when committed to land. This tends to make the resultant data conservative since slightly better performance could be achieved in full beep. For actual single-engine failures, the beep control should be used for maximum capability of the operating engine. The crew should also be aware that rotor overspeed can occur with full beep when the thrust control rod is lowered after landing.

20. The operator's manual suggests regaining normal operating rotor speed following a single-engine failure. At gross weights of 40,000 to 46,000 pounds, normal operating rotor speed is 245 rpm. The increased reduction in collective pitch to regain 245 rpm versus 235 rpm caused an initial increase in sink rate which resulted in approximately 10 percent more height loss to reach the target airspeed. Because of these results, the touchdown tests for the CH-47C were conducted using 235-rpm rotor speed following the simulated failure. When an immediate landing is required following a single-engine failure, the pilot should regain 235 rotor rpm for all gross weights. The operator's manual suggestion to regain normal operating rotor speed is adequate when continued flight is possible.

21. The CH-47C is susceptible to the phenomenon of power settling, often referred to as the vortex ring state. This condition was encountered following the simulation of single-engine failures at an out-of-ground-effect (OGE) hover in very light wind conditions. When the thrust control rod was lowered simultaneously with or slightly before the nose-down pitching of the helicopter, the settling resulted. Power settling was characterized by a high rate of descent in a hover attitude. Forward cyclic control was initially ineffective, which prevented an increase in airspeed. The application of a large amount of forward cyclic control during power settling caused a slight nose-up pitching which aggravated the settling condition. Recovery from this condition was achieved by further lowering the thrust control rod until the cyclic control was effectively able to pitch the helicopter to an accelerating attitude. This recovery required approximately twice the amount of altitude normally lost and, depending on the initial hover height, could result in ground contact. This condition was observed at conditions where single-engine OGE hover capability did not exist. The helicopter was more susceptible to power settling with a slight tail wind than with a head wind. To avoid power settling following a simulated single-engine failure from an OGE hover, it was necessary to pitch to the accelerating attitude immediately after the 2-second delay time and prior to lowering the thrust control rod. When lowering the thrust control rod, a nose-up pitching moment occurred that varied with the rate of thrust control rod application. Figure 4, appendix F, shows the magnitude of the pitch-up with a moderate thrust control rod rate (1 inch per second (in./sec)) and the amount of forward cyclic required to control the pitching. The pitch-up was minimized by lowering the thrust control rod at a slower rate (approximately 1/2 in./sec). Extensive pilot compensation was required to control pitch attitude following a simulated single-engine failure from an OGE hover (HQRS 6). This is a shortcoming and should be corrected for improved safety of operation. A discussion of the power-settling phenomenon and the technique used to prevent it should be incorporated in the operator's manual as shown below:

CAUTION

Power settling can result if the thrust control rod is lowered first, following a single-engine failure from an out-of-ground-effect hover. The helicopter should first be pitched to an accelerating attitude before the thrust control rod is slowly lowered (1/2 inch per second) to regain rotor speed. If power settling is encountered, the recovery may require twice the amount of altitude normally lost. The helicopter is more susceptible to power settling following an engine failure while hovering with a slight tail wind.

22. The following discussion of pilot technique following single-engine failure should be included in the operator's manual:

> Following determination that an engine failure has occurred, the pilot should immediately lower the nose of the helicopter to an accelerating attitude prior to lowering the thrust control rod, if the airspeed is slow and altitude permits. If the speed at the time of the failure is near hover, the accelerating attitude should be between 15 and 20 degrees nose down and should be reached with a relatively rapid pitch rate (approximately 16 degrees per second (deg/sec)). For airspeeds between 30 knots indicated airspeed (KIAS) and approximately 50 KIAS (52 KIAS), the accelerating attitude should be between 10 and 15 degrees and should be reached with a moderate pitch rate (approximately 9 deg/sec). For airspeeds between approximately 50 and 75 KIAS (52 and 74 KIAS), the accelerating attitude should be between 5 and 10 degrees and should be reached with a slow-to-moderate pitch rate (approximately 7 deg/sec). Steeper pitch attitudes and faster pitch rates will improve performance but may be uncomfortable to the pilot. While the pilot is assuming the accelerating attitude, the copilot should advance the normal engine trim control switch to full beep to gain maximum available power on the operating engine. After the helicopter is established in an accelerating attitude, the thrust control rod should be slowly lowered (1/2 in./sec) as necessary to regain the desired rotor speed. If the thrust control rod is lowered too rapidly, a nose-up pitching will occur that will delay the airspeed build. If flight conditions are such that continued flight with one engine operable is possible, normal operating rotor speed of either 235 rpm or 245 rpm should be regained. The performance section of the operator's manual contains information on best single-engine operation. When an immediate landing is required following an engine failure, the pilot should establish rotor speed at 235 rpm for all gross weights. If conditions in excess of 50 KIAS at a minimum of 100 feet above the ground in

the accelerating attitude are reached, a safe running landing is possible for all operational gross weights, assuming the terrain is satisfactory. If these conditions of airspeed and altitude are not met or landing terrain is unsuitable, some damage to the helicopter should be expected. Upon completion of the landing and before lowering the thrust control rod, the normal engine trim control switch must be reduced to the governing range to prevent rotor overspeed. For conditions of airspeed greater than approximately 75 KIAS (74 KIAS) and height at or above 30 feet above the ground, the helicopter should be smoothly decelerated to touchdown speeds of between 20 and 30 knots.

AIRSPEED/ALTITUDE LANDING WINDOW

23. Recognizing that operational pilots would use varying flare techniques, it was necessary to develop H-V diagrams which are compatible with these techniques. Accordingly, final flare and touchdown tests were accomplished using various flare rates and starting at various flare altitudes and airspeeds. The results of these tests show that at an airspeed of 58 KCAS and a 100-foot height AGL, with the aircraft in the accelerating attitude appropriate for the entry airspeed (para 17), safe run-on landings at touchdown speeds of approximately 30 knots could be made using a wide range of flare rates and heights. Slow flare rates beginning at approximately 75 feet and faster flare rates beginning at approximately 45 feet were equally successful. This airspeed/altitude window (58 KCAS/100 feet AGL) was used to define the H-V diagrams. When the window conditions were reached, an end point was established for defining that particular point of the H-V diagram. For the conditions tested, this technique resulted in similar levels of pilot compensation. However, higher density altitude or lower power-available conditions than encountered during this test may require a greater degree of pilot compensation and a subsequent change to the window parameters.

DENSITY ALTITUDE EFFECTS

24. Density altitude effects on H-V performance were evaluated at the conditions shown in table 3. The minimum entry heights were obtained using the window concept (para 23). The data show that for the same gross weight, an increase in density altitude resulted in a greater height loss. Density altitude effects were more pronounced as entry airspeed was decreased and the maximum effect occurred at the high hover (OGE) point. No apparent density altitude effect was noted at the low hover (in-ground-effect (IGE)) point and from this point no difference in touchdown technique was required for the density altitudes tested.

Table 3. Density Altitude Comparison.[1]

Average Density Altitude (ft)	Average Gross Weight (lb)	Calibrated Entry Airspeed (kt)	Average Outside Air Temperature (°C)	Minimum Entry Height (ft)
4500	40,910	Hover	-2.5	460
650	40,870	Hover	-1.0	390
4500	40,910	43	-2.5	155
650	40,870	42	-1.0	150
4500	44,110	43	-2.0	178
-200	44,080	43	-5.5	140
4030	46,100	42	-9.0	192
900	46,050	39	+4.5	155

[1]Entry rotor speed: 245 rpm.
Average center of gravity: FS 327 (mid).

GROSS WEIGHT AND CENTER-OF-GRAVITY EFFECTS

25. Gross weight effects were evaluated at nominal gross weights of 40,800, 44,000, and 46,000 pounds at density altitudes near sea level and approximately 4500 feet. The entries were made from stabilized level flight and the maneuver continued to a touchdown. The minimum entry heights were obtained using the window concept (para 23) and are presented in table 4. At the higher density altitude, an increase in gross weight resulted in a greater height loss. At the lower density altitude, H-V performance remained approximately the same for the gross weights tested. Gross weight had minimal effect on H-V performance at the lower density altitude because of the proximity of the entry altitude to the window altitude.

Table 4. Gross Weight Effects.[1]

Average Gross Weight (lb)	Average Density Altitude (ft)	Calibrated Entry Airspeed (kt)	Average Outside Air Temperature (°C)	Minimum Entry Height (ft)
40,910	4500	43	-2.5	155
44,110	4500	43	-2.0	178
46,100	4030	43	-9.0	192
40,870	650	42	-1.0	150
44,080	-200	43	-5.5	140
46,050	900	39	+4.5	155

[1]Entry rotor speed: 245 rpm.
Average center of gravity: FS 327 (mid).

26. Gross weight effects in a hover were evaluated IGE only, since an OGE hover capability did not exist at the higher gross weights. The IGE tests were conducted from a stabilized hover, and following the simulated single-engine failure, a vertical descent to a touchdown was accomplished. The maximum safe height was determined based on pilot qualitative comments, minimum transient rotor speed, or the height from which a running landing could be made. At 40,800 pounds and a 650-foot density altitude, a running landing could be made from 30 feet. Gross weight had a significant effect on the maximum safe IGE hover height, as shown in table 5. At no time did the gear loads approach limit values.

Table 5. In-Ground-Effect Hover Results.[1]

Average Gross Weight (lb)	Average Density Altitude (ft)	Average Outside Air Temperature (°C)	Maximum Safe In-Ground-Effect Height (ft)	Minimum Transient Rotor Speed (rpm)	Maximum Gear Load[2] (lb)
40,800	650	-1.0	31	209	12,305
44,000	-200	-3.0	26	193	11,212
46,000	400	-3.0	19	181	Note[3]
44,000	5200	2.0	20	192	15,007
46,000	5200	1.5	15	188	15,180

[1]Entry rotor speed: 245 rpm.
Average center of gravity: FS 327 (mid).
[2]Gear loads listed for the critical parameter (aft gear spindle housing).
Maximum allowable load is 18,000.
[3]Data not available.

27. During the development of the pilot technique, cg locations at FS 319.5 (forward), FS 324.0 (mid), and FS 334.5 (aft) were evaluated at a 38,000-pound gross weight. For these conditions, no handling qualities or performance differences were detected by the pilots following simulated single-engine failures. Subsequent landing tests were accomplished at a mid cg.

HEIGHT-VELOCITY PERFORMANCE PREDICTION

28. An attempt was made to predict H-V performance using the analytical procedures developed by USAASTA during CH-47B H-V testing (ref 12, app A). It was possible to match the analytical performance with data flown by tailoring the forcing functions to shape the results. With changes in flight conditions, corresponding changes in the state variables (gross weight, density altitude, etc.) did not result in accurate predication of H-V performance without appropriate change in the forcing functions. The forcing-function changes could not be determined before the testing was completed. Currently, additional work is being accomplished to develop an improved analytical method for H-V prediction. This analytical method should be available for test application in the near future.

29. A further attempt to predict H-V performance was made using the referred-gross-weight method. This method envisioned a generalization of data based on the ratio of gross weight to density ratio (W/σ). The data collected both at altitude and during touchdown testing did not generalize. The failure to generalize can be attributed to the difference in power available on the operating engine (due to ambient temperature differences) and compressibility effects at the different conditions even though W/σ remained constant.

30. The inability to predict H-V performance using the above methods led to H-V profiles being defined at constant gross weights at several density altitudes, which increased the amount of flight testing originally anticipated and also increased the risk of the entire test program. In addition, test results could only be obtained at the test conditions available.

AIRSPEED CALIBRATION

31. The ship airspeed system was calibrated in the slow-speed range using the boom airspeed system as a standard. The results are shown in figure 5, appendix F, for gross weights above 40,000 pounds. Between 50 and 108 KCAS, the error was as presented in the operator's manual (ref 1, app A). Below 50 KCAS, the error deviates up to 2 KCAS from the data shown in reference 1. The difference can be attributed to the heavier weights and resultant increase in downwash. For these conditions of gross weight, a minimum reliable airspeed indication of 30 KIAS (41 KCAS) was determined and is incorporated in the operational H-V diagrams presented. The airspeed calibration developed during this test was used to obtain indicated airspeed for the operational presentation.

OPERATIONAL SINGLE-ENGINE HEIGHT-VELOCITY PERFORMANCE

Out-of-Ground-Effect Hover and Level Flight

32. Single-engine H-V diagrams were determined by stabilizing the helicopter in level flight at the desired conditions of airspeed and height above the ground, then placing one engine condition lever in the ground position and accomplishing a landing. The techniques established during this test (paras 16 through 23) were used in determining the minimum heights required to accomplish a safe landing. Nominal gross weights of 40,800, 44,000, and 46,000 pounds were tested to a touchdown at the conditions shown in figures 6 and 7, appendix F. At airspeeds of 58 KCAS and slower, the entry height at each airspeed was determined by incrementally decreasing height above the touchdown point until the window conditions of 58 KCAS at 100 feet could no longer be achieved. This determined the minimum height AGL required for a safe landing following a single-engine failure. A smooth deceleration was used at entry airspeeds from 59 to 79 KCAS. At entry airspeeds greater than 79 KCAS, the window conditions do not apply, 100 feet of altitude was no longer required to accomplish a safe landing, and a minimum safe height of 30 feet was chosen. Tests were conducted at speeds in excess of 100 KIAS as low as 20 feet above the ground, to demonstrate the capability for a safe landing following a single-engine failure. At all speeds, the helicopter reaction was a slight nose-up pitching which precluded any tendency to abruptly settle into the ground.

33. The H-V diagrams presented in figures 6 and 7, appendix F, were defined at relatively low outside air temperatures, which affected maximum power available. Further testing would be required at similar gross weight and density altitudes, but at higher outside air temperatures to more completely define the single-engine H-V performance of the CH-47C helicopter equipped with T55-L-11A engines. Operational single-engine H-V diagrams (figs. 8 and 9) were developed from the conditions shown in figures 6 and 7, using the airspeed calibration discussed in paragraph 31. For the conditions tested, no OGE hover capability existed except at a gross weight of 40,800 pounds. The operational single-engine H-V diagrams developed during this test are suitable for presentation in the operator's manual when accompanied by gross weight, density altitude, and outside air temperature information and a discussion of the recommended pilot technique (para 22).

In-Ground-Effect Hover and Takeoff

34. The CH-47C was tested during IGE hover at gross weights to 46,000 pounds and density altitudes of 650 and 5200 feet (para 26, table 5). A safe vertical landing from a hover with no ground roll was made from 20 feet at 44,000 pounds, and from 15 feet at 46,000 pounds.

35. The takeoff from a hover for the CH-47C is described very generally in the operator's manual (ref 1, app A). It advises the pilot to increase airspeed and altitude simultaneously after reaching translational lift. The maneuver is started from a stabilized hover height of 10 feet. The CH-47 standardization guide

published by the US Army Aviation School (ref 13) is more explicit. It calls for accelerating from a 10-foot hover to translational lift in a level attitude, not exceeding 20 feet until reaching 30 KIAS and thereafter simultaneously gaining altitude and airspeed. Testing was conducted to verify the safety of the recommended takeoff techniques in the event of a single-engine failure. The data, presented in figure 10, appendix F, were obtained at gross weights of 40,800 and 46,000 pounds at density altitudes near 1000 feet. The recommended takeoff technique was used except that pitch attitudes up to 5 degrees, nose low, were tested. To remain below 20 feet and not exceed a 5-degree nose-low pitch attitude while accelerating to 30 KIAS, engine torque had to be limited to approximately 5 percent above that required for a 10-foot hover. The pilot technique used after the simulated failure was to level the aircraft, set the thrust control rod to regain 235 rpm rotor speed (if time permitted), and to complete a run-on landing. Beep trim was unchanged in this test. In all cases tested, including a steep climb after reaching 30 KIAS, a safe landing could be made using normal roll-on procedures. While at low altitude (below 20 feet) it is necessary to avoid rapid aft cyclic movement to prevent the aft gear from contacting the ground. The normal takeoff procedure described in the operator's manual and the CH-47 standardization guide is safe in the event of single-engine failure. The procedure can be safely expanded to include an attitude of 5 degrees, nose low, during takeoff. This takeoff technique is recommended for training and operations, to reduce the possibility of damage following a single-engine failure.

Landing Approach

36. The operator's manual (ref 1, app A) has a description of procedures for the pilot to follow for a single-engine approach, but does not discuss approaches from which a safe landing can be made in the event of a single-engine failure. The CH-47 standardization guide (ref 13) does explain the procedures for shallow (5- to 8-degree), normal (8- to 10-degree), and steep (12- to 15-degree) approaches, but without regard to the degree of risk in the event of a single-engine failure during the approach.

37. The CH-47C helicopter was tested during approaches at gross weights of 40,800 and 46,000 pounds at the conditions listed in table 6. The pilot technique after the simulated failure was to maintain the airspeed until the landing flare, set the thrust control rod to regain 235-rpm rotor speed (if time permitted), and to complete the run-on landing. No reduction in thrust control rod position was made unless adequate altitude remained. The beep trim was unchanged in this test to preclude rotor overspeed on touchdown. This procedure was necessary because of the extra crew workload during the testing, which is not present operationally. Full beep is recommended in the event of actual engine failure.

Table 6. Approach Test Conditions.[1]

Average Gross Weight (lb)	Type Approach (deg)	Average Density Altitude (ft)	Average Outside Air Temperature (°C)	Entry Height Above Ground Level (ft)	Calibrated Airspeed At Failure (kt)	Entry Rate Of Descent (ft/min)
40,800	Normal (8 to 10)	900	-0.5	22	[2]5	300
				28	26	378
				37	31	450
				53	43	432
				72	36	468
				78	46	582
				125	55	624
46,000	Shallow (5 to 6)	1580	8.0	15	[2]4	180
				24	20	246
				35	37	324
				36	39	360
				38	43	486
				56	41	558
				63	46	450
46,000	Normal (8 to 10)	1200	3.5	81	50	552
				84	52	564
				101	63	690

[1]Entry rotor speed: 245 rpm.
Average center of gravity: FS 327 (mid).
[2]Estimated airspeed.

38. At a 40,800-pound gross weight and the conditions in table 6, safe landings were made following simulated single-engine failures on a normal (8- to 10-degree) approach without undue pilot effort, using the normal run-on landing technique. Qualitatively, it was determined that simulated single-engine failures from a ste.. approach at gross weights of 40,800 pounds and above would result in hard landings and were not tested. A discussion of the risk involved in approaches, such as the note below, should be included in the operator's manual and the CH-47 standardization guide.

NOTE

> At a 40,800-pound gross weight or greater, an engine failure during the final portion of a steep approach may result in a hard landing.

39. At a 46,000-pound gross weight ..d the conditions in table 6, safe landings following a simulated single-engine failure were made from normal (8- to 10-degree) approaches when failure occurred at or above 41 KIAS on the ship's airspeed system (50 KCAS). Qualitatively, it was determined that simulated failures below this airspeed, during a normal approach, would result in hard landings and were not tested. Simulated single-engine failures were successfully conducted at a 46,000-pound gross weight using a shallow (5- to 6-degree) approach. However, simulated failures below 30 KIAS on the ship's airspeed system (41 KCAS) required large thrust control rod adjustments and resulted in a minimum transient rotor speed below 200 rpm. These points are very near maximum performance for these flight conditions. A discussion of the risk involved in approaches, such as the note below, should be included in the operator's manual and the CH-47 standardization guide.

NOTE

> At a 46,000-pound gross weight, an engine failure during the final portion of a normal approach may result in a hard landing.

CONCLUSIONS

GENERAL

40. The following conclusions were reached upon completion of the height-velocity tests of the CH-47C helicopter equipped with T55-L-11A engines:

a. Single-engine height-velocity performance was similar for similar conditions regardless of which engine remained operational (para 13).

b. Within the scope of this test, the entry characteristics following simulated single-engine failure are satisfactory (para 14).

c. Power settling following single-engine failure from an out-of-ground-effect hover can be avoided by immediately pitching the helicopter to the accelerating attitude and then slowly lowering the thrust control rod to regain rotor speed (para 21).

d. A safe running landing can be made following a single-engine failure if the aircraft can be accelerated to 58 knots calibrated airspeed prior to reaching 100 feet above ground level (para 23).

e. Where conditions of gross weight and outside air temperature are similar, an increase in density altitude resulted in a greater height loss (para 24).

f. At high density altitudes and similar outside air temperatures, an increase in gross weight resulted in a greater height loss (para 25).

g. Gross weight had a significant effect on the height-velocity performance following a simulated single-engine failure from an in-ground-effect hover (para 26).

h. There were no detectable changes in height-velocity performance due to center-of-gravity location (para 27).

i. Attempts to analytically predict and generalize height-velocity performance were unsuccessful (paras 28 and 29).

j. The minimum reliable airspeed indication is 30 knots indicated airspeed (41 knots calibrated airspeed) (para 31).

k. The operational single-engine height-velocity diagrams developed during this test are suitable for presentation in the operator's manual when accompanied by gross weight, density altitude, outside air temperature information, and a discussion of the recommended pilot technique (para 33).

l. The takeoff procedures described in the operator's manual and the CH-47 standardization guide are safe in the event of a single-engine failure and can be safely expanded to include an attitude of 5 degrees, nose low (para 35).

m. At a 40,800-pound gross weight or greater, an engine failure during the final portion of a steep approach may result in a hard landing (para 38).

n. At a 46,000-pound gross weight, an engine failure during the final portion of a normal approach may result in a hard landing (para 39).

o. No deficiencies and one shortcoming were identified during this test.

SHORTCOMING AFFECTING MISSION ACCOMPLISHMENT

41. Correction of the shortcoming, extensive pilot compensation required to control pitch attitude following a simulated single-engine failure from an out-of-ground-effect hover, is desirable (HQRS 6) (para 21).

RECOMMENDATIONS

42. The shortcoming, correction of which is desirable, should be corrected.

43. The following information should be included in the operator's manual:

a. A "CAUTION" with discussion of the power-settling phenomenon and the technique used to prevent it (para 21).

b. The pilot technique following single-engine failure (para 22).

c. The operational height-velocity diagrams accompanied by gross weight, density altitude, and outside air temperature information, and a discussion of the recommended pilot technique (para 33).

d. A steep-approach NOTE: "At a 40,800-pound gross weight or greater, an engine failure during the final portion of a steep approach may result in a hard landing." (para 38).

e. A normal-approach NOTE: "At a 46,000-pound gross weight, an engine failure during the final portion of a normal approach may result in a hard landing." (para 39).

44. The normal takeoff technique, described in the operator's manual and CH-47 standardization guide, should be used for operations and training, whenever possible, to reduce the possibility of damage following single-engine failure (para 35).

APPENDIX A. REFERENCES

1. Technical Manual, TM 55-1520-227-10, *Operator's Manual, Army Model CH-47B and CH-47C Helicopters*, 5 August 1970, with changes through 26 October 1971.

2. Report, The Boeing Company, Vertol Division, Number 114-FT-030-1, *Analyses of CH-47B Helicopter Height-Velocity Testing Program.*

3. Final Report, USAASTA, Project No. 68-02, *CH-47B Height-Velocity Evaluation*, February 1970.

4. Letter, AMSAV-GR(R-F), AVSCOM, 3 November 1969, subject: Request for Test, CH-47C Height Velocity.

5. Test Plan, USAASTA, Project No. 69-17, *CH-47C Height-Velocity*, September 1971.

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

7. Specification, The Boeing Company, Vertol Division, Number 114-PJ-803, *Detail Specification for the Model CH-47C Helicopter*, 7 December 1967, with Revision A, 15 May 1967.

8. Technical Manual, TM 55-1520-227-20, *Organizational Maintenance Manual, Army Model CH-47B and CH-47C Helicopters*, 6 August 1970, with changes through 18 August 1971.

9. Message, AVSCOM, AMSAV-EF, 08-16, 25 August 1971, Unclas, subject: CH-47C Height Velocity Test.

10. Manual, USAASTA, "Helicopter Performance Testing Guidebook," unpublished.

11. Final Report, USAASTA, Project No. 66-29, *Airworthiness and Flight Characteristics Test, CH-47C Helicopter (Chinook), Stability and Control*, March 1972.

12. Technical Note No. 16, USAASTA, *Discussion of the Hoffman Autorotation Performance Model*, 22 June 1970.

13. Standardization Guide, US Army Aviation School, Fort Rucker, Alabama, *Standardization of Helicopter Maneuvers, CH-47*, July 1970.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The test CH-47C helicopter was equipped with two Lycoming turboshaft T55-L-11A engines mounted in separate nacelles on the aft fuselage. The engines (each rated at 3750 shaft horsepower, sea level, standard day) drive two three-bladed rotors in tandem through a combining transmission, drive shafting, and reduction transmissions. A gas turbine hydraulic auxiliary power unit drives the aft transmission accessory gearbox to provide hydraulic and electrical power for engine starting and other ground operations when the rotors are stopped. A pod containing three fuel tanks is located on each side of the fuselage. The helicopter is equipped with fixed landing gear. An entrance door is located at the forward right side of the cabin fuselage section. A hydraulically powered loading ramp is located at the rear of the cargo compartment. The pilot seat and controls are located on the right side of the cockpit; the copilot seat and controls are located on the left side.

Physical Dimensions

Length (fuselage)	51.0 ft
Length (rotors turning)	99.0 ft
Overall height (rotors stationary)	18.7 ft
Width of cabin	9.0 ft
Tread (forward gear)	10.5 ft
Tread (aft gear)	11.2 ft
Rotor diameter	60.0 ft
Rotor solidity	0.067
Number of rotors	2
Blades per rotor	3
Disc area (total)	5655 ft^2
Swept area	5000 ft^2 (approx)

Weight Data

Empty weight (specification)	20,420 lb
Design gross weight	33,000 lb
Alternate design gross weight	46,000 lb

Operational Rotor Speeds

Gross weight of 40,000 pounds or less	235 rpm
All gross weights (normally used only above 40,000 pounds)	245 rpm

CENTER-OF-GRAVITY LIMITATIONS

Forward Limit

2. The extreme forward limit is FS 301 up to a gross weight of 28,550 pounds. From this point, the forward cg limit decreases linearly to FS 309.7 at a gross weight of 33,000 pounds. From this point, the forward cg limit again decreases linearly to FS 319.7 at a gross weight of 46,000 pounds.

Aft Limit

3. The extreme aft cg limit is FS 349 up to a gross weight of 28,550 pounds. From this point, the aft cg limit decreases linearly to FS 338 at a gross weight of 33,000 pounds. From this point, the aft cg limit decreases to FS 335 at a gross weight of 46,000 pounds.

Fuselage Station

4. The fuselage station is measured in inches from the reference datum line located 21.5 inches forward of the nose of the helicopter.

APPENDIX C. DATA ANALYSIS PROCEDURES

DATA CONSISTENCY

1. To ensure data cr. ation and technique consistency, three critical parameters were identified and allowable deviation established. These parameters and limits are as follows:

   a. Pitch attitude, ±3 degrees.

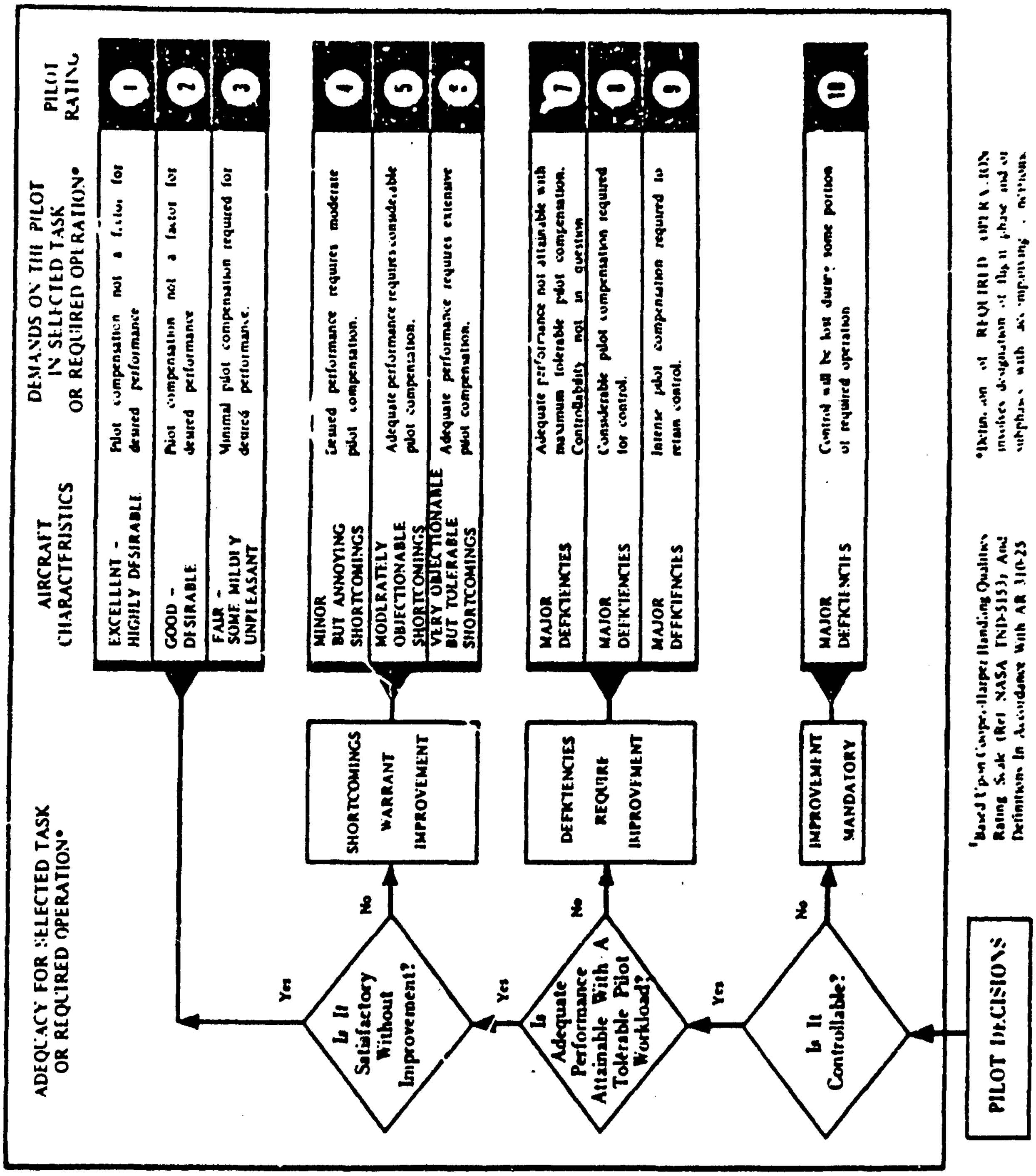
   b. Pitch rate, ±3 degrees per second.

   c. Delay time, ±0.3 second.

EXCEPTIONS TO CONVENTIONAL HEIGHT-VELOCITY TEST TECHNIQUES

2. The Fairchild Flight Analyzer was not used for data acquisition during this test. At Tonopah Test Range, Nevada, the primary data acquisition method used was radar space positioning, with the AN/APN 171 radar altimeter as a secondary method. While at Tonopah, the radar altimeter was calibrated by space positioning. At Edwards Air Force Base, California, the radar altimeter was used as the primary data acquisition method. The radar altimeter, with altitude information on both the photopanel and the oscillograph, was adequate for data acquisition for the test technique used. Altitude and rate-of-descent information was available from the oscillograph trace.

APPENDIX D. HANDLING QUALITIES RATING SCALE



APPENDIX E. TEST INSTRUMENTATATION

COCKPIT PANEL

Boom airspeed
Ship's system airspeed
Rotor speed
Boom altitude
Ship's system altimeter
Angle of sideslip
Angle of attack
Longitudinal control position
Lateral control position
Directional control position
Thrust control rod (collective control) position
Cruise guide indicator
Radar altimeter indicator

PHOTOPANEL

Boom airspeed
Ship's system airspeed
Rotor speed
Gas producer speed (N_1) (both engines)
Boom altitude
Ship's system altitude
Free air temperature
Fuel temperature (both engines)
Fuel used (both engines)
Engine torque (both engines)
Rate of climb/descent
Time of day
Correlation counter
Camera counter
Oscillograph record counter (No. 1 and No. 2)
Event light (pilot)
Event light (engineer)

OSCILLOGRAPH NO. 1

Rotor blip
Engine fuel flow (both engines)

Aft pivoting-link actuator
Aft fixed-link actuator
Cruise guide indicator
Forward gear oleo extension (left and right)
Aft gear oleo extension (left and right)
Aft gear shock axial load (left and right)
Aft gear upper drag load (left and right)
Aft gear axial load spindle (left and right)
Aft gear lower drag bending (left and right)
Aft gear vertical acceleration (left and right)
Aft gear touchdown switch (left and right)
Voltage monitor
Photopanel camera blip
Engineer event
Pilot event

OSCILLOGRAPH NO. 2

Boom airspeed
Longitudinal control position
Lateral control position
Directional control position
Thrust control rod (collective control) position
Differential collective pitch (DCP) speed trim position
Forward cyclic speed trim position
Throttle position (both engines)
Pitch SAS (both channels) (No. 1 and No. 2)
Roll SAS (both channels)
Yaw SAS (both channels)
Pitch attitude
Roll attitude
Yaw attitude
Pitch rate
Roll rate
Yaw rate
Pitch acceleration
Roll acceleration
Yaw acceleration
Angle of attack
Angle of sideslip
Gas producer speed (N_1) (both engines)
Center-of-gravity normal acceleration

Rotor speed
Rotor blip
Radar altitude
Photopanel camera blip
Engineer event
Pilot event

APPENDIX F. TEST DATA

INDEX

FIGURE [illegible]

MINIMUM TRANSIENT ROTOR SPEED FOLLOWING SINGLE ENGINE FAILURE

CH-47C USA S/N 68-15834

ENGINES: T55-L-11A

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG C.G. (IN)	ENTRY ROTOR SPEED (RPM)	ENTRY THRUST COEF [illegible]	ENTRY FLIGHT CONDITION
○	40910	4800	+2.5	333.8(AFT)	245	52.75	LEVEL
○	44110	4600	-2.0	326.2(MID)	245	45.37	LEVEL
○	45100	3070	+4.0	327.3(AFT)	245	65.25	LEVEL
■	40810	850	+1.0	325.8(MID)	245	51.32	LEVEL
■	44080	-200	+8.5	326.2(MID)	245	65.54	LEVEL
◆	40010	920	+8	321.3(AFT)	245	60.57	LEVEL

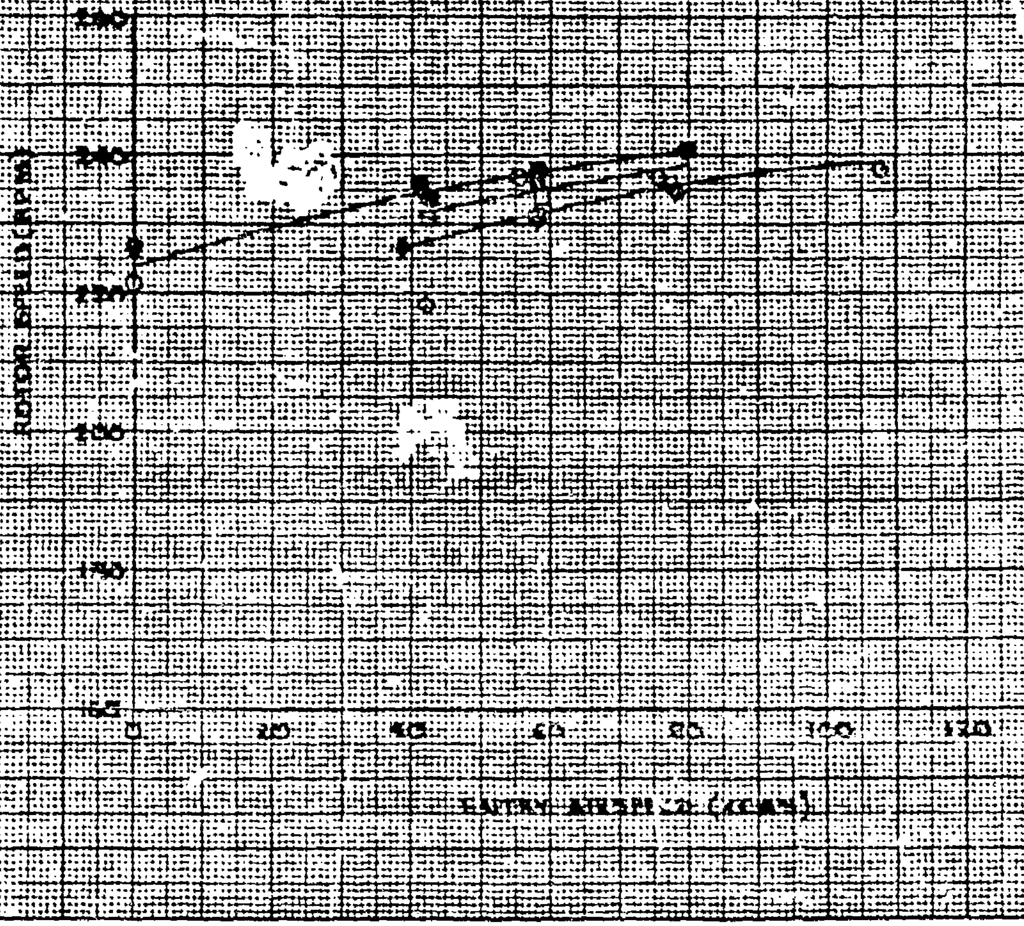
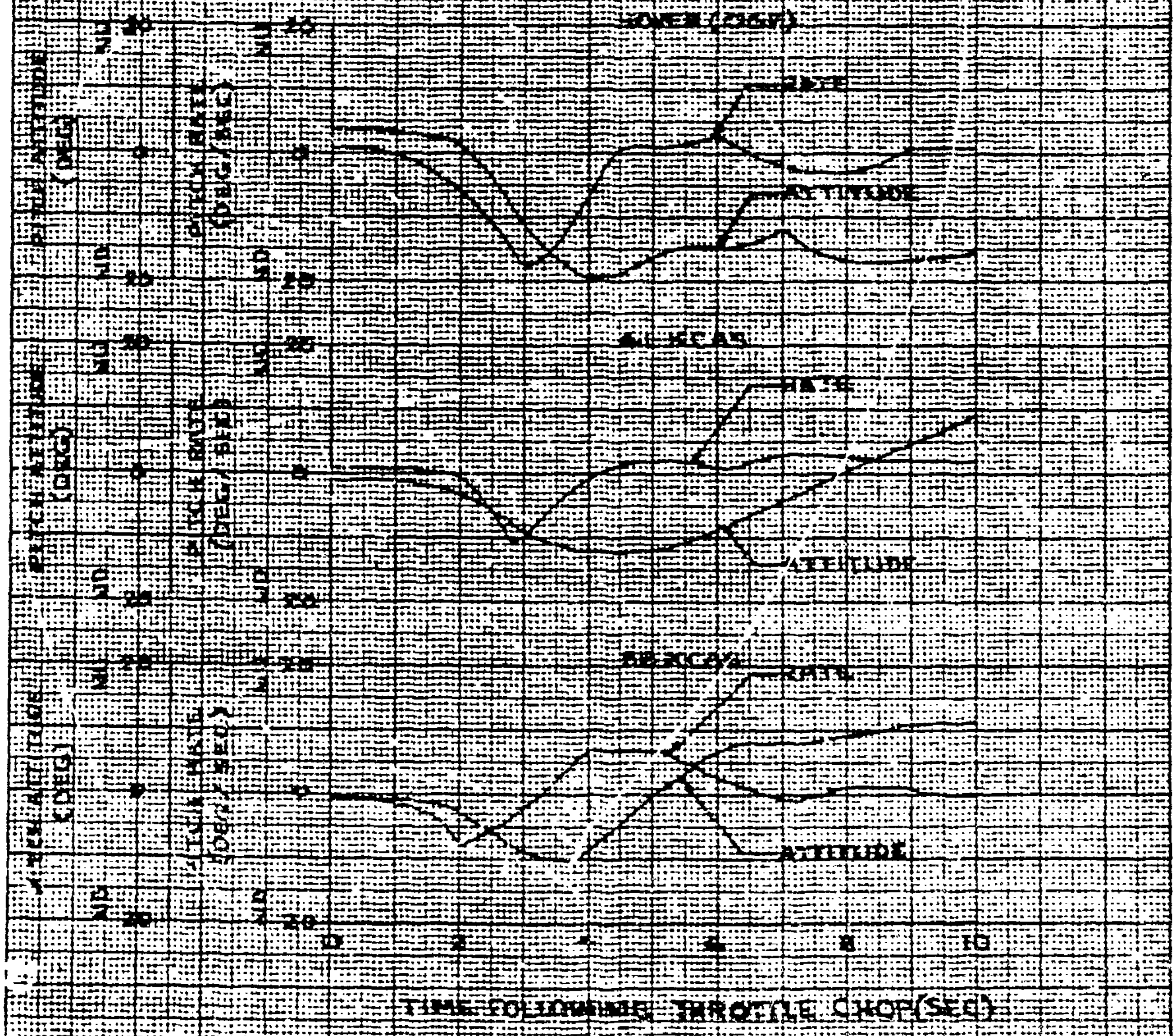
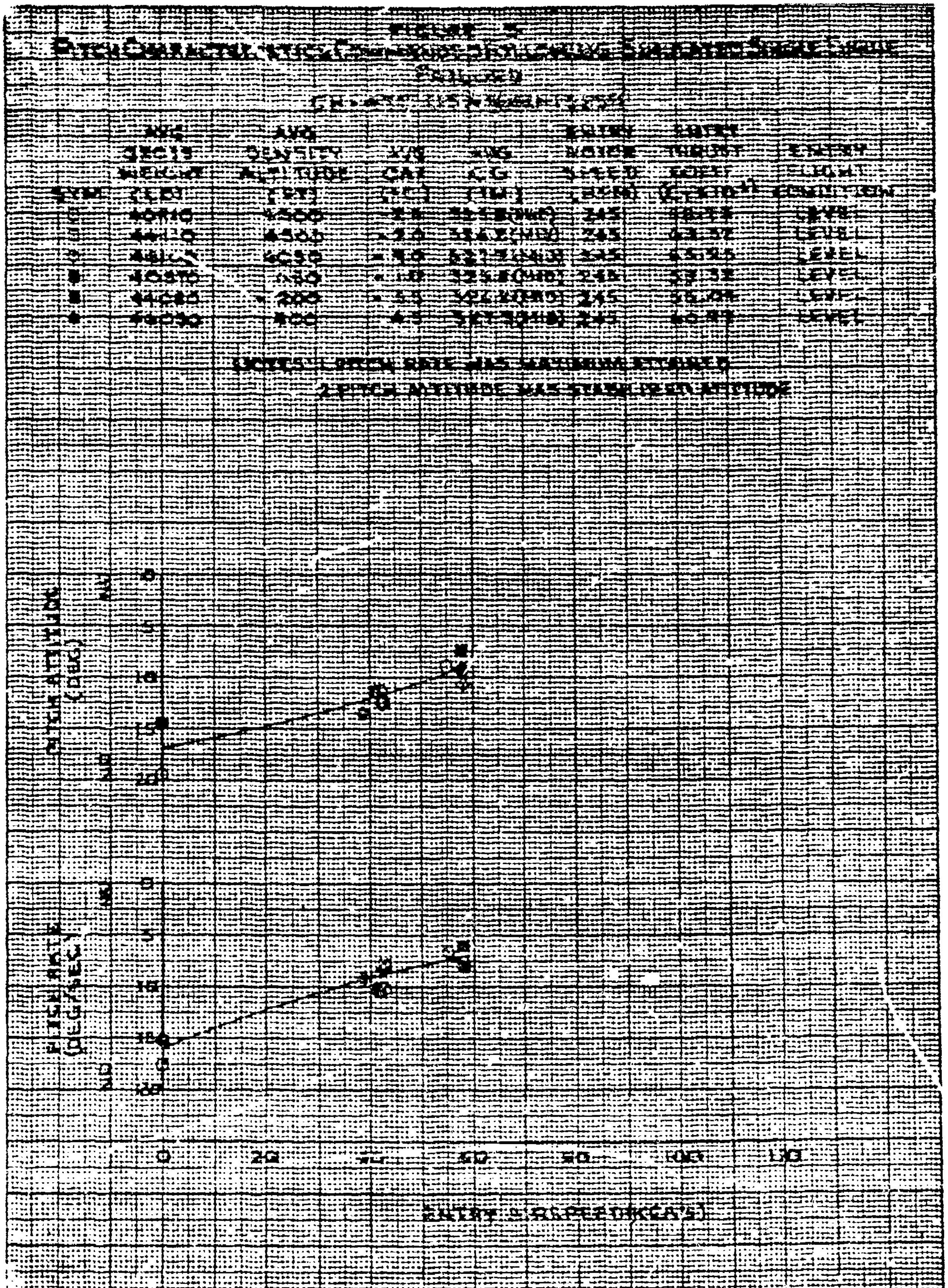


FIGURE 1

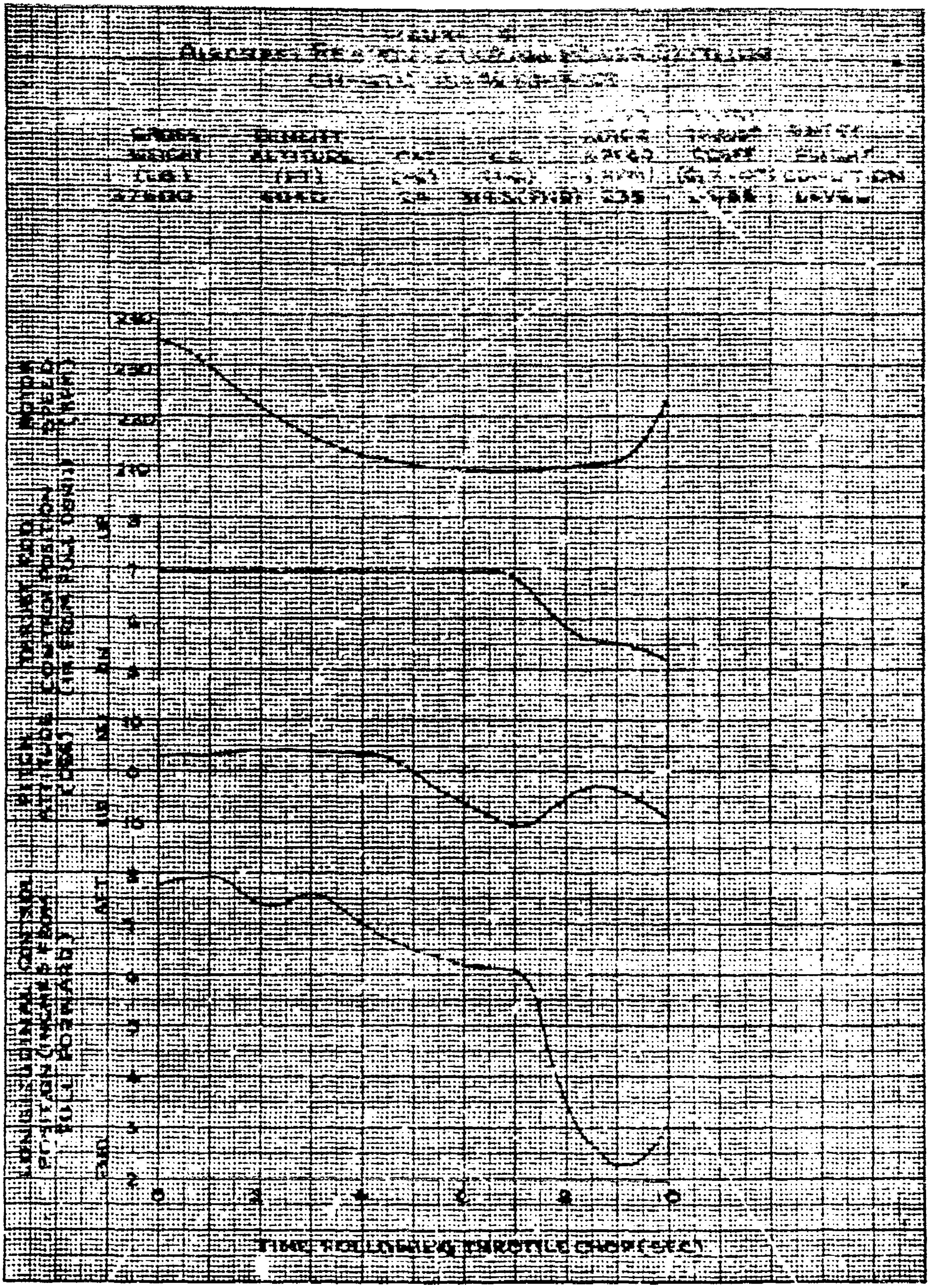
PITCH CHARACTERISTICS COMPARISONS FOLLOWING SIMULATED SINGLE ENGINE FAILURES

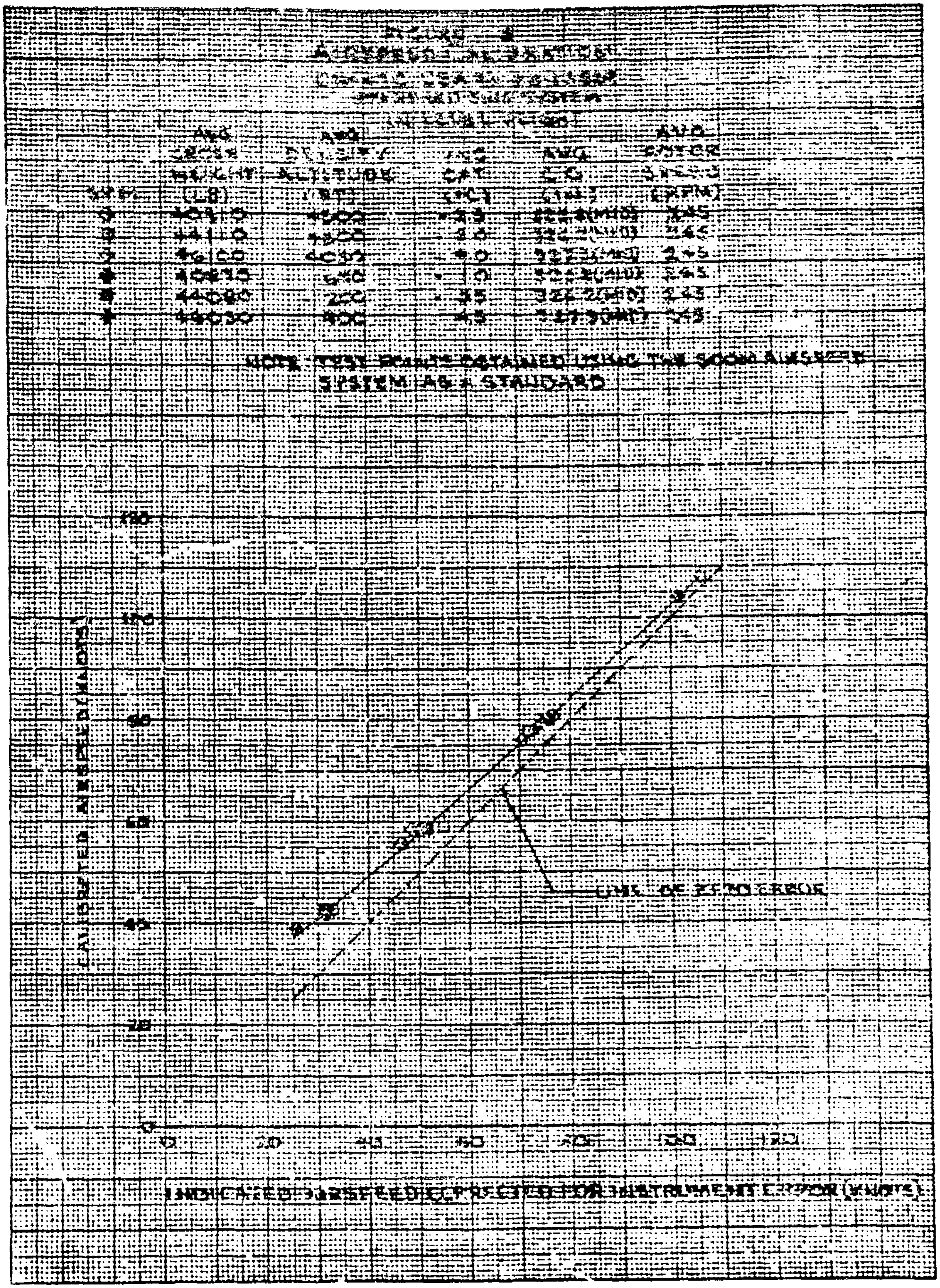
AVG GROSS WEIGHT (LB)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG CG (IN)	ENTRY ROTOR SPEED (RPM)	ENTRY THRUST COEF	ENTRY FLIGHT CONDITION
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	LEVEL





AVG
DENSITY
ALTITUDE
CAT
C.G
ENTRY
SPEED
THRUST
ENTRY
CONDITION
SYM
245
LEVEL
PITCH ATTITUDE (DEG)
PITCH RATE (DEG/SEC)
0
20
40
60
80
100
120
ENTRY AIRSPEED (KCAS)


TIME FOLLOWING THROTTLE CHOP (SEC)


SYSTEM AS A STANDARD
LINE OF ZERO ERROR
INDICATED AIRSPEED CORRECTED FOR INSTRUMENT ERROR (KNOTS)
CALIBRATED AIRSPEED (KNOTS)

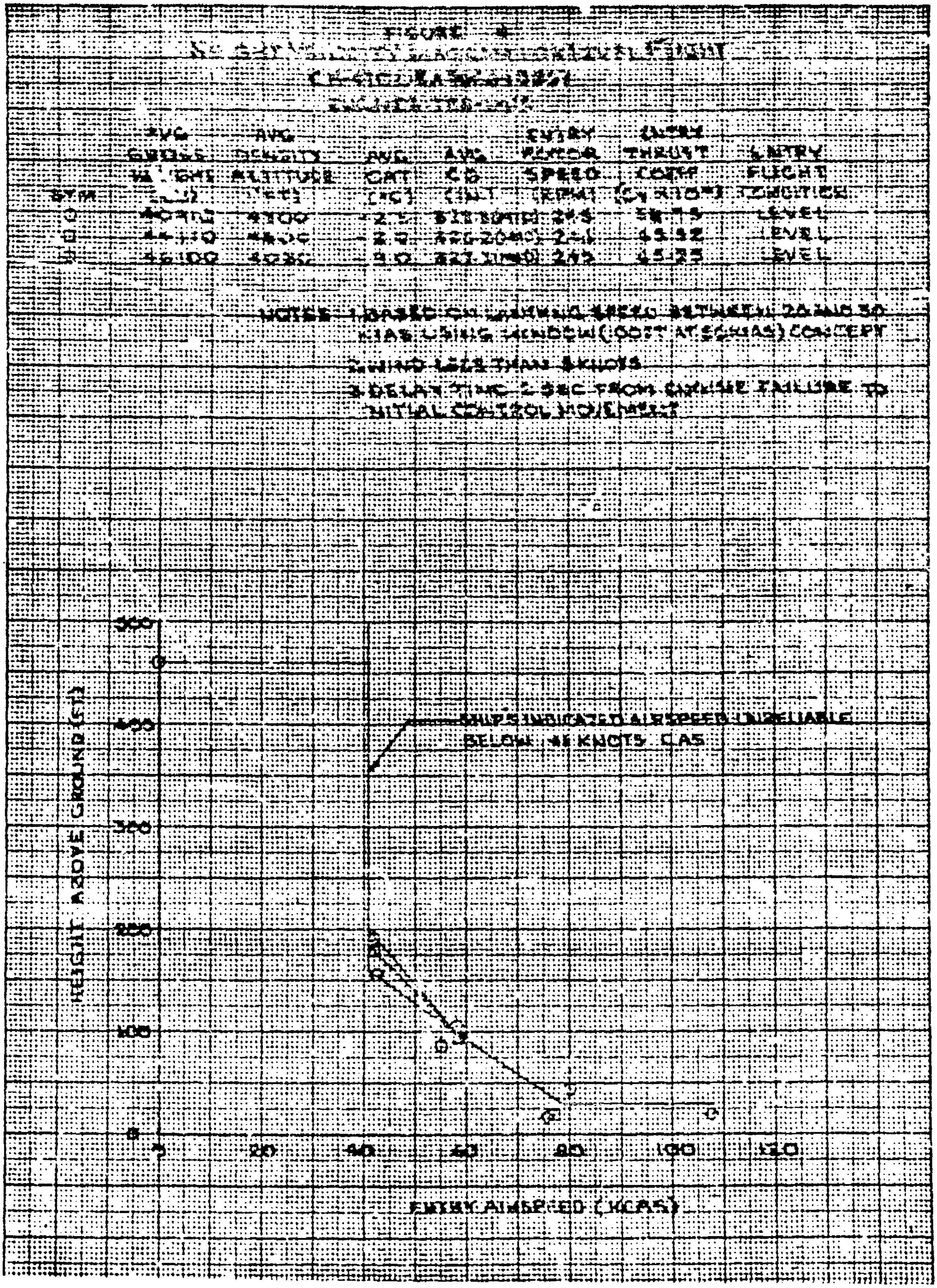
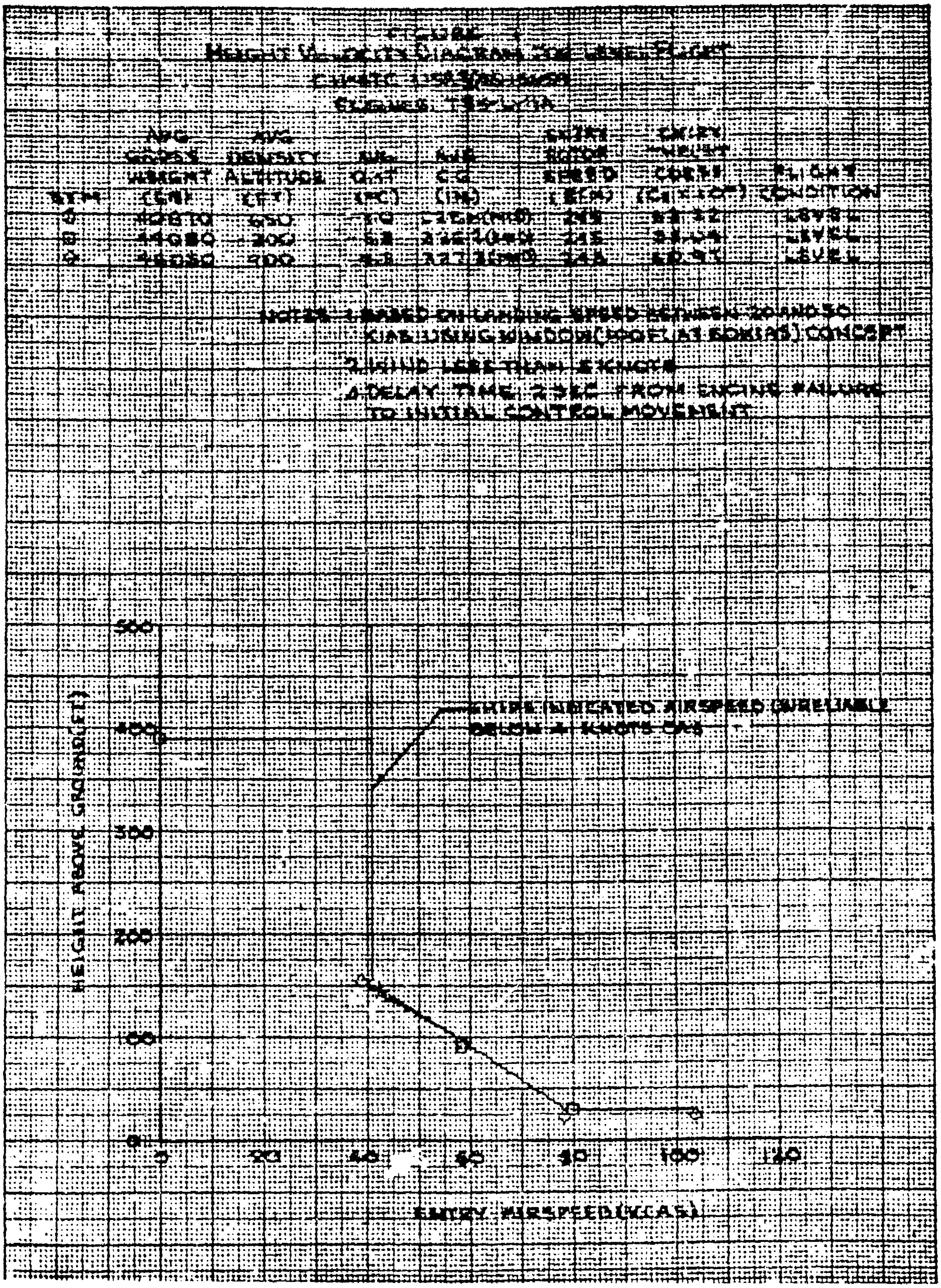


FIGURE
AVG
AVG
ENTRY
ENTRY
GROSS
DENSITY
AVG
AVG
ROTOR
THRUST
ENTRY
WEIGHT
ALTITUDE
OAT
CG
SPEED
COEFF
FLIGHT
SYM
CONDITION
LEVEL
LEVEL
LEVEL
NOTES
BELOW
KNOTS CAS
500
400
300
200
100
0
20
40
60
80
100
120
ENTRY AIRSPEED (KCAS)
HEIGHT ABOVE GROUND (FT)



ALTITUDE
CONDITION
DELAY TIME 2 SEC FROM ENGINE FAILURE
TO INITIAL CONTROL MOVEMENT
500
400
300
200
100
0
HEIGHT ABOVE GROUND (FT)
0
20
40
60
80
100
120
ENTRY AIRSPEED (KCAS)

FIGURE 8

OPERATIONAL HEIGHT-VELOCITY DIAGRAM FOR LEVEL FLIGHT

CH-47C USA S/N [illegible]

ENGINE T55-L-11A

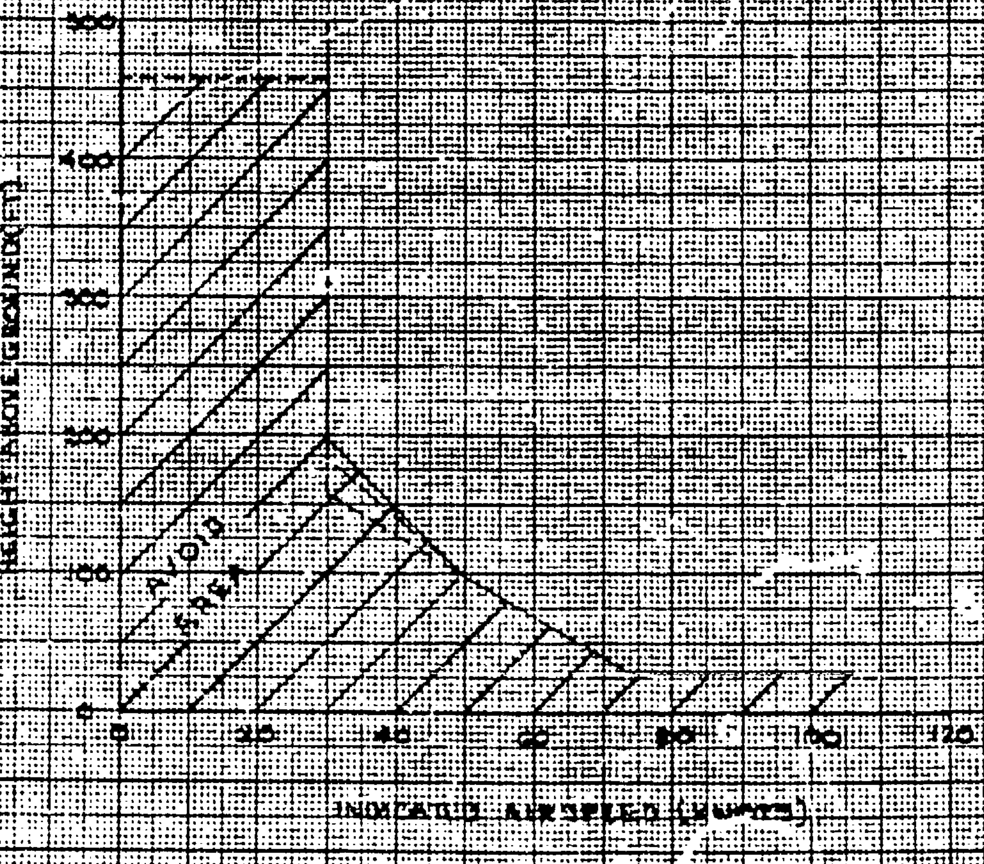
SYM	GROSS WEIGHT (LB)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITION
	41000	4500	+25	245	LEVEL
	44000	4200	+20	245	LEVEL
	46000	4000	+10	245	LEVEL

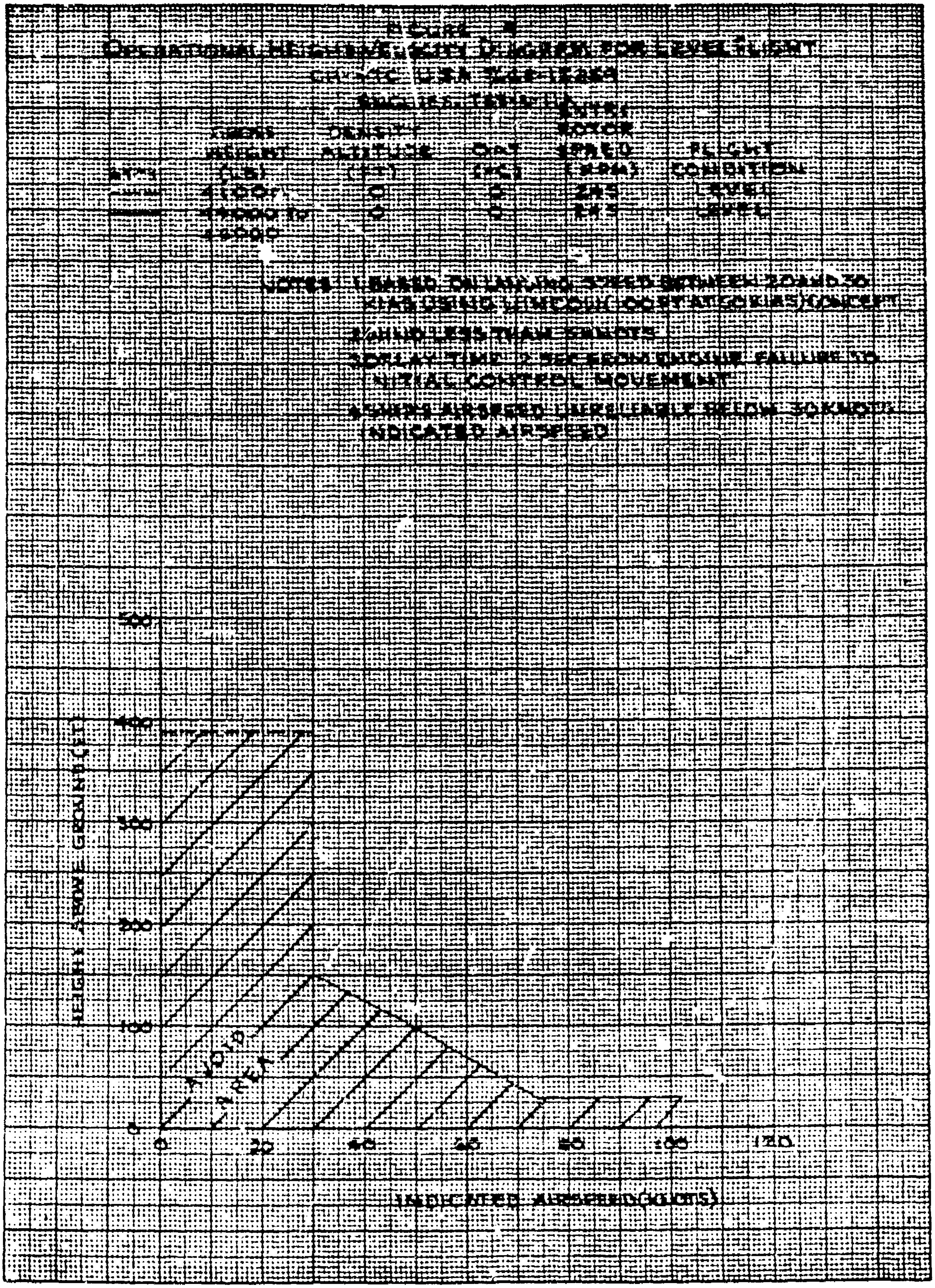
NOTES: 1. BASED ON LANDING SPEED BETWEEN 20 AND 30 KIAS USING [illegible] AT 30 KIAS CONCEPT

2. WIND LESS THAN 5 KNOTS

3. DELAY TIME 2 SEC FROM ENGINE FAILURE TO INITIAL CONTROL MOVEMENT

4. SHIPS AIRSPEED UNRELIABLE BELOW 30 KNOTS INDICATED AIRSPEED





OPERATIONAL HEIGHT-VELOCITY DIAGRAM FOR LEVEL FLIGHT
GROSS WEIGHT (LB)
DENSITY ALTITUDE (FT)
OAT (°C)
ROTOR SPEED (RPM)
FLIGHT CONDITION
LEVEL
LEVEL
NOTES:
INITIAL CONTROL MOVEMENT
INDICATED AIRSPEED
HEIGHT ABOVE GROUND (FT)
500
400
300
200
100
0
AVOID AREA
0
20
40
60
80
100
120
INDICATED AIRSPEED (KNOTS)

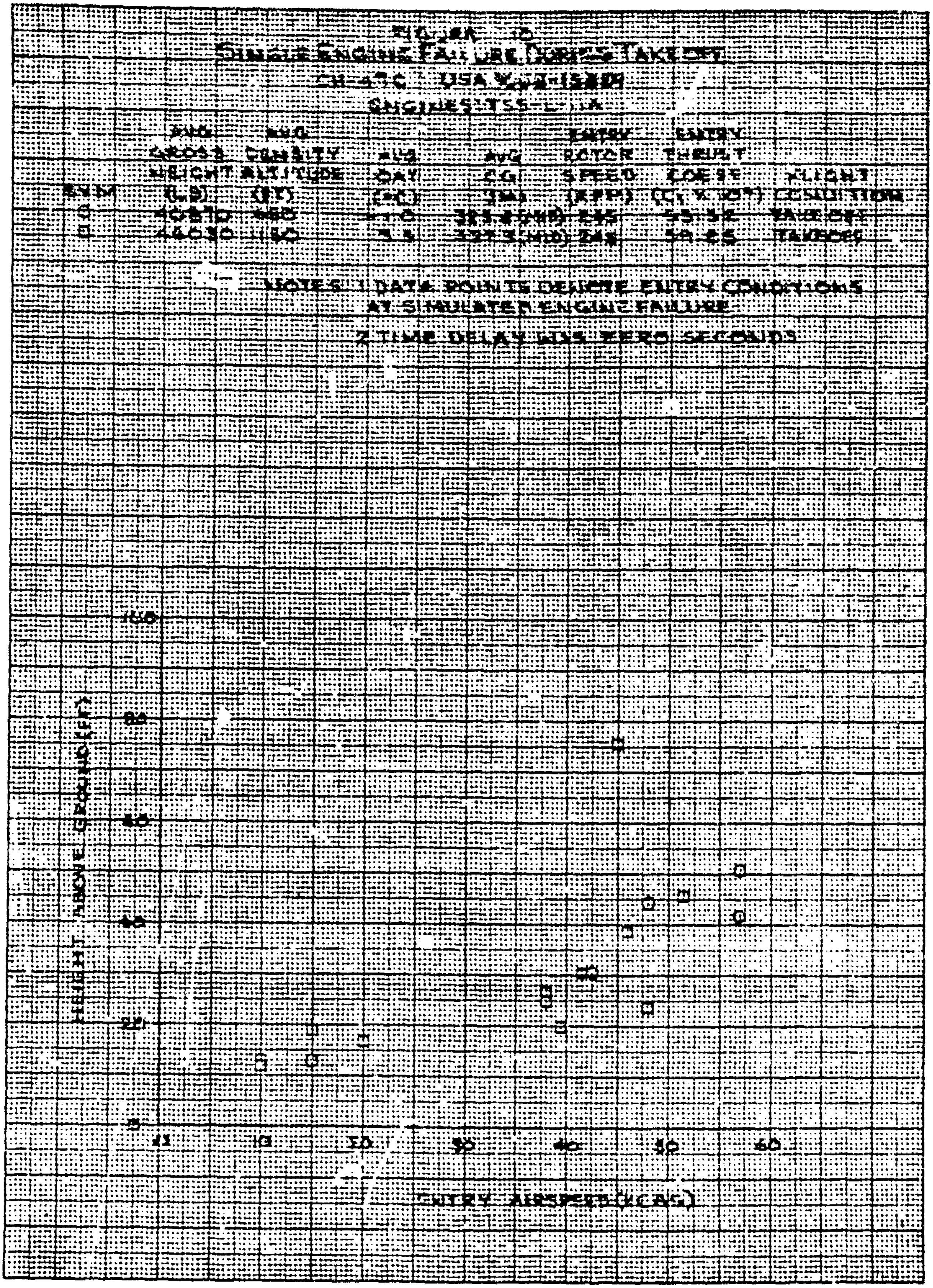


FIGURE 10
SINGLE ENGINE FAILURE DURING TAKEOFF
AVG
GROSS
WEIGHT
(LB)
AVG
DENSITY
ALTITUDE
(FT)
AVG
CG
ENTRY
ROTOR
SPEED
(RPM)
ENTRY
THRUST
COEFF
FLIGHT
CONDITION
NOTES: 1 DATA POINTS DEFINE ENTRY CONDITIONS AT SIMULATED ENGINE FAILURE
2 TIME DELAY WAS ZERO SECONDS
HEIGHT ABOVE GROUND (FT)
100
80
60
40
20
0
10
20
30
40
50
60
ENTRY AIRSPEED (KCAS)

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13. ABSTRACT

The CH-47C height-velocity flight test program was conducted at Edwards Air Force Base and Shafter, California, and Tonopah Test Range, Nevada, between 29 September 1971 and 9 March 1972. Engineering flight tests were conducted to develop realistic single-engine height-velocity diagrams for the CH-47C helicopter with T55-L-11A engines. During these tests, no deficiencies were identified, but one shortcoming was identified: the excessive pilot compensation required to control pitch attitude following a simulated single-engine failure from an out-of-ground-effect hover. The height-velocity diagrams developed are suitable for inclusion in the operator's manual when accompanied by the flight conditions and a discussion of the pilot technique. Entry characteristics of the helicopter following engine failure are satisfactory. Power settling may occur following an engine failure from an out-of-ground-effect hover unless the helicopter is pitched immediately to an accelerating attitude before the thrust control rod is lowered. The takeoff procedures depicted in the operator's manual and the US Army Aviation School CH-47 standardization guide are safe in the event of single-engine failure. However, hard landings may result when a power failure occurs during a steep approach at or above a 40,800-pound gross weight or during a normal approach at a 46,000-pound gross weight. Increases in gross weight and density altitude degraded height-velocity performance. Efforts to generalize height-velocity performance data using analytical procedures and referred-gross-weight methods were unsuccessful. Height-velocity performance was apparently unaffected by the center-of-gravity location or which engine was failed. Further testing at high outside air temperatures would be required to completely define the single-engine height-velocity performance of the CH-47C helicopter equipped with T55-L-11A engines.

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