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AN EVALUATION OF THE HEIGHT VELOCITY DIAGRAM OF A HEAVYWEIGHT, HIGH ROTOR INERTIA, SINGLE ENGINE HELICOPTER

TECHNICAL REPORT

NOVEMBER 1966

by
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National Aviation Facilities Experimental Center

FEDERAL AVIATION AGENCY
AIRCRAFT DEVELOPMENT SERVICE
Washington, D. C.

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ABSTRACT

A series of flight tests was conducted at three selected altitudes (sea level, 5000 feet, 6500 feet) to determine the effects of altitude and weight on the height-velocity (H-V) diagram of a large, heavyweight, high rotor inertia, high disk loading, single rotor, single engine helicopter. Three gross weights of the helicopter were used. Quantitative and qualitative test data were collected to determine how the H V diagram varies with density altitude and aircraft gross weight. An investigation was made into the effects on the diagram of a delayed collective pitch application response.

Results disclosed a family of curves showing that increases in density altitude and/or gross weight enlarged the H-V diagram required for a safe power-off landing. Analysis of the results revealed that the key points (V_{cr}, h_{min}, and h_{max}), which partially define the curves, could be determined by the solution of a set of linear equations. These results were identical to those reported in FAA Technical Reports ADS-1 and ADS-46 except for the constants of the linear equations and the location of the critical height (h_{cr}). The critical height indicated a slight increase as weight, altitude and collective pitch reduction time delay were increased. An average value for h_{cr} can be selected without upsetting the family of curves.

TABLE OF CONTENTS

	Page
ABSTRACT	i
INTRODUCTION	1
Purpose	1
Background	1
DISCUSSION	1
Test Aircraft	1
Test Instrumentation	2
Test Operations and Procedures	2
Flight Test Sites	2
Test Methodology	2
Test Criteria	6
ANALYSIS AND RESULTS	8
Discussion of Tests	8
Height-Velocity Diagrams	9
Discussion of One-Second Delay	19
Effects of Weight and Altitude	23
Equations	23
Constant H-V Diagram for Reduction of Weight With Altitude	27
CONCLUSIONS	29
REFERENCES AND BIBLIOGRAPHY	30
ACKNOWLEDGEMENTS	31
APPENDIX 1 Glossary of Terms (1 page)	1-1
APPENDIX 2 Test Aircraft Specifications and Instrumentation Details (6 pages)	2-1
APPENDIX 3 Summary of Pilot's Comments (7 pages)	3-1
APPENDIX 4 Summary of Height-Velocity Diagram Flight Test Data (2 pages)	4-1

LIST OF ILLUSTRATIONS

Figure		Page
1	Typical Height-Velocity Diagram	2
2	Test Aircraft	3
3	Typical Test Site Layout	5
4	Height-Velocity Diagrams - Basic Data. Helicopter Gross Weight, 9100 pounds. Three Density Altitudes Shown	10
5	Height-Velocity Diagrams - Basic Data. Helicopter Gross Weight, 10,100 pounds. Three Density Altitudes Shown	11
6	Height-Velocity Diagrams - Basic Data. Helicopter Gross Weight, 11,100 pounds. Two Density Altitudes Shown	12
7	Height-Velocity Diagram Variation With Density Altitude and Gross Weight Delay, No Delay and All Conditions Shown	13
8	Height-Velocity Diagram Variation With Density Altitude. Gross Weights of 9100, 10,100, and 11,100 pounds. Delay and No Delay Conditions Shown	14
9	Height-Velocity Diagram Variation With Gross Weight. Three Density Altitudes Shown for Delay and No Delay Conditions	15
10	Comparison of Time History Data for High Hover Points	20
11	Comparison of Time History Data for Critical Speed Area (V_{cr}, h_{cr})	21
12	Comparison of Time History Data for Low Hover Points	22
13	Critical Velocity (V_{cr}) Versus Aircraft Gross Weight for the Range of Test Density Altitudes	24

LIST OF ILLUSTRATIONS CONTINUED

Figure		Page
14	Critical Velocity (V_{cr}) Versus Test Altitude for the Range of Test Weights	24
15	Low Hover Height (h_{max}) Versus Aircraft Gross Weight for the Range of Test Density Altitudes	24
16	Low Hover Height (h_{max}) Versus Test Altitude for the Range of Test Weights	25
17	High Hover Height (h_{min}) Versus Square of Critical Velocity (V^2_{cr})	26
18	Constant H-V Diagram - Weight Reduction	28
2-1	Airborne Instrumentation (Interior)	2-4
2-2	Airborne Instrumentation (Exterior)	2-5
2-3	Space Positioning Equipment	2-6
2-4	Data Center and Meteorological Equipment	2-6

LIST OF TABLES

Table		Page
I	Summary of High Hover (h_{min}) and Near High Hover Data	17
II	Summary of Typical Data-Area of Critical Speed (V_{cr}) and Critical Height (h_{cr})	18

INTRODUCTION

Purpose

The purpose of this project was to determine by flight tests the effects of altitude and weight on the height-velocity (H-V) diagrams of a large single-rotor helicopter which has an inherently high rotor inertia and high disk loading.

Background

This flight test project is the culmination of a program initiated by the Aircraft Development Service, Federal Aviation Agency, to acquire sufficient actual flight test data on certain basic helicopter flight parameters associated with the determination of the H-V diagram. The ultimate objective of this program is to obtain a practical technical approach for the determination of the effects of altitude and aircraft gross weight on the helicopter H-V diagram.

The H-V diagram is a chart which defines an envelope of flight with respect to airspeed and height above the ground where, in the event of power failure, a safe power-off landing could not be accomplished. A typical H-V diagram as referred to in this report is shown in Fig. 1. It is a diagram established from data based on the criteria of steady-state, level-flight entry conditions.

Previous flight test projects of this program are reported in References 1 and 2. The flight test data obtained on these projects disclosed that the H-V diagrams of the lightweight helicopters tested resolved into a family of curves as a function of weight and altitude. To further confirm this relationship it was felt advisable to examine the autorotative characteristics of a heavyweight single-rotor helicopter for further correlation.

The helicopter utilized for the tests reported herein generally represents the high extreme in the spectrum of current generation single-engine helicopters with respect to considerations of gross weight, disk loading and rotor inertia.

DISCUSSION

Test Aircraft

The test vehicle was a large, heavyweight, single-rotor, single engine helicopter as shown in Fig. 2. This aircraft was selected for this H-V test project because of its relatively high rotor inertia and high disk loading. Pertinent specifications of this aircraft are presented in Appendix 2.

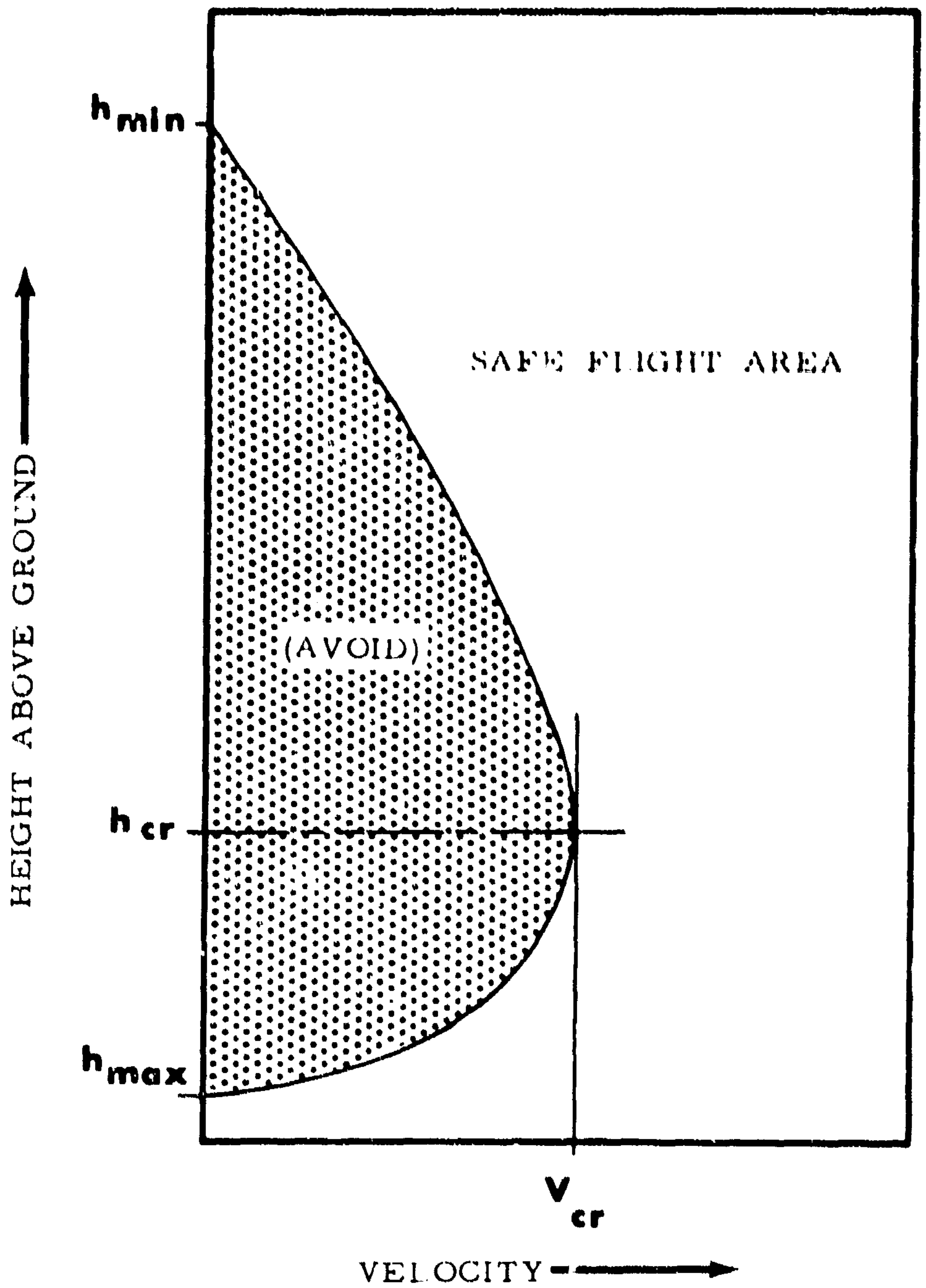

FIG. 1 TYPICAL HEIGHT-VELOCITY DIAGRAM

FIG. 2 TEST AIRCRAFT

Test Instrumentation

Airborne and ground instrumentation was utilized to record helicopter performance and meteorological data. Details of the quantitative information measured and the equipment utilized are presented in Appendix 2.

Test operations and Procedures

1. Flight Test Sites

The flight test project was conducted at test sites in the State of California during the period September 1965 through February 1966. The test sites, selected for their elevation and test environments, were as follows:

Thermal Airport	Elevation -117 ft. MSL
Bishop Airport	Elevation 4118 ft. MSL
Lake Tahoe Airport	Elevation 6263 ft. MSL

A schematic view of the test site layout showing the relative location of the test course, space positioning equipment, meteorological equipment and the test control center is shown in Fig. 3.

2. Test Methodology

A professional engineering test pilot well skilled in the mechanics of determining H-V diagrams was employed for the piloting tasks. The results of his airwork are therefore representative of flight skills beyond the realm of average pilot capabilities and consequently produced minimum size H-V diagrams.

A total of 1044 test runs were conducted to determine H-V diagrams at the selected test altitudes for gross weight conditions of 9100 pounds, 10,100 pounds, and 11,100 pounds.

The following is a general description of the manner in which the tests were conducted:

a. General

The pilot would fly over the test course at a specific steady airspeed at a predetermined entry height above the ground. When stabilized, he would execute a simulated power failure by sudden retardation of the throttle in order to fully disengage the rotor clutch. From this point he would land the aircraft with the power off. This procedure was repeated with the pilot adjusting his height or airspeed until he reached a point below which he felt a safe landing could not be made because all usable energy had been expended. This point was then plotted as a point on the H-V diagram.

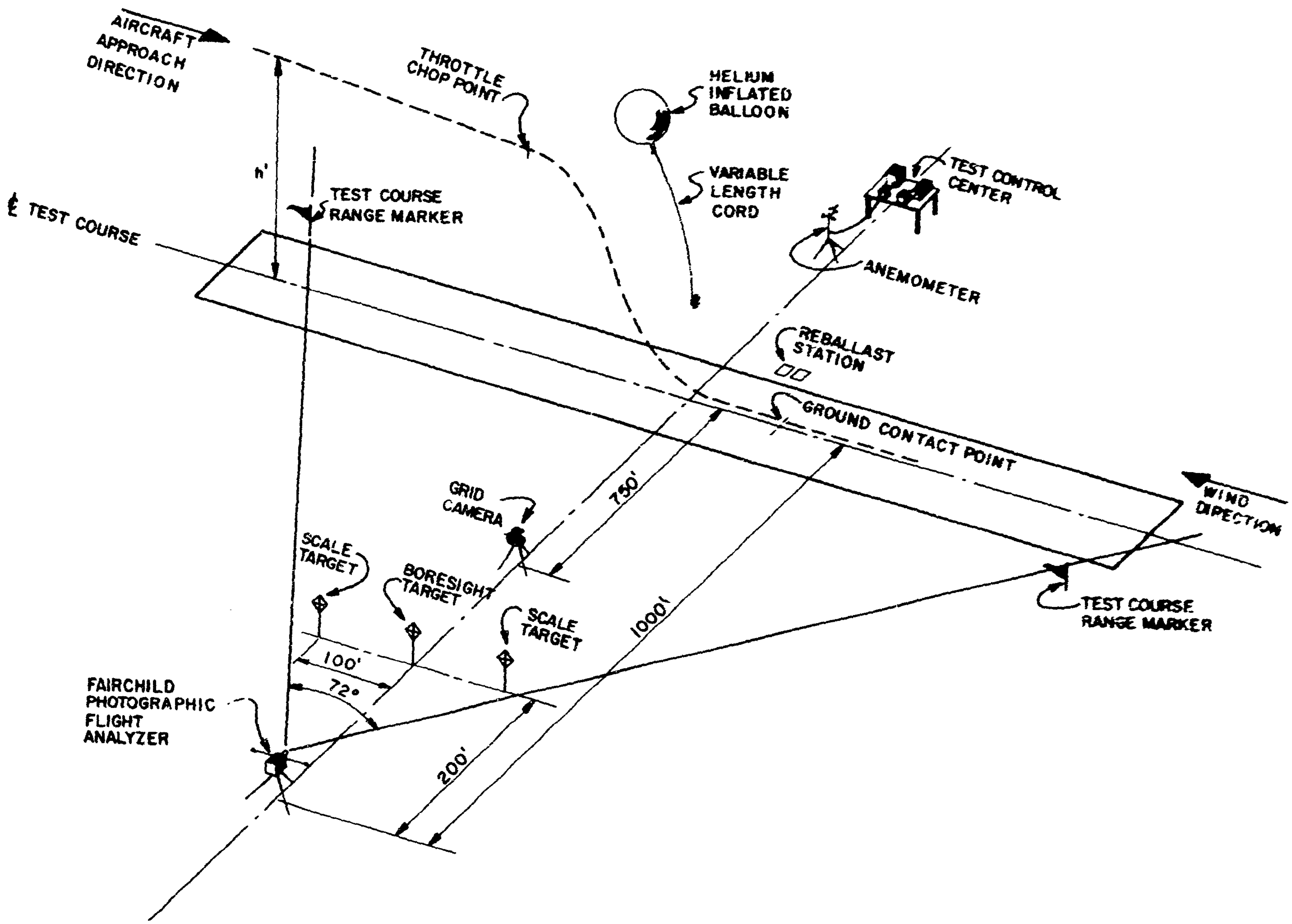

FIG. 3 TYPICAL TEST SITE LAYOUT

The validity of his judgment was checked by means of limited on-site data reduction to determine if the point thus declared was usable as a valid data point.

The above procedure was repeated until a sufficiency of points was obtained from which an H-V diagram could be generated.

b. Collective Pitch Control Application

The usual procedure when power fails in flight with a single engine helicopter is for the pilot to retain the highest possible rotor speed to effect a landing. This is accomplished by immediate full reduction of the rotor blade pitch angle by means of the collective pitch stick control when the height above the ground is adequate. When the height above the ground and the consequent time differential between power failure and touchdown is limited, it is not always possible to effect full collective pitch reductions. In such cases, the pilot makes partial collective pitch reductions or simply utilizes what collective pitch he has remaining as the situation dictates.

The fact that the test vehicle had inherently high rotor inertia suggested a comparative investigation into the effects of a no-delay and one-second delayed response in reducing collective pitch following throttle cut in order to correlate these results with those findings reported in Reference 2. Since the one-second delay data reported in Reference 2 indicated that a displacement or step at the "knee" of the curve carried on up to the high hover height, it was felt necessary to determine whether a high rotor inertia rotor would exhibit the same characteristics as compared to a low rotor inertia system. Tests using a one-second delay response with collective pitch application were therefore programmed into the test plan.

3. Test Criteria

a. Rotor Speed

In order to eliminate as many variables as possible, the rotor speed in steady state autorotation was kept constant at a given weight by adjusting the low pitch blade angle at each altitude tested. This involved raising the low pitch setting slightly at each increase in test altitude by changing the length of the pitch link. Total collective pitch travel, therefore, was always available for control purposes.

b. Pilot Procedures

There were no restrictions placed on horizontal touchdown velocity; that is, the pilot was not instructed to obtain minimum touchdown speed, nor was he limited as to his maximum touchdown speed.

The specific piloting techniques for handling the helicopter were left to the discretion of the pilot. The only limitations in technique imposed upon the pilot were that of the no-delay and one-second delay in collective pitch reduction after throttle cut.

The decision as to whether a landing was a maximum performance effort was made by the pilot. His evaluation was based on whether he believed he had any usable reserve energy remaining in the form of rotor speed or airspeed, and the nature and magnitude of the impact (Pilot estimated landing load factors). The pilot's qualitative comments on techniques utilized and the related criteria for his decisions were used in evaluating the flight test data. A discussion of these techniques can be found under "Pilot's Comments" in Appendix 2.

c. Weight and Center of Gravity Control

Weight was kept within approximately ± 1/2 percent by adding ballast after every few runs and refueling as required.

The Center of Gravity (c.g.) of the helicopter was generally constant for all tests at a location one inch (± 1/2 inch) forward of the vertical station through the rotor hub.

d. Wind Allowables

Limitations were placed on allowable wind velocities for these tests. The wind velocities were measured at a 12 foot instrumentation height. Hovering and very slow speed tests were not conducted in wind velocities in excess of 2 mph, and all other tests were discontinued when the wind exceeded 5 mph at this height. A helium filled balloon moored so its height could be varied was utilized as a visual indicator of wind aloft for the benefit of the pilot.

e. Altitude Control

Density altitude for the tests, with but a few exceptions, was maintained within approximately 600 feet of the average density altitude for each consideration of weight and collective application technique. The exceptions were evaluated and weighed in the final determination of the H-V curves. It was considered that small variations in density altitude would have little effect on the test data results.

f. Entry Speeds and Conditions

All speeds used in the program and in this report are given in terms of calibrated airspeed (CAS). The entry airspeed used for each point on the H-V diagram was obtained from the photographic record as ground speed, corrected for observed wind at the 12 foot level and converted to calibrated airspeed.

ANALYSIS AND RESULTS

Discussion of Tests

A brief discussion of several aspects of the test program at this point would perhaps contribute to a better understanding of the test results. The test vehicle, which was large and heavy, was not generally sensitive to the effects of light, steady winds, particularly in the very low speed regimes. Problems due to wind did exist however, because of inability to accurately determine the winds aloft which could vary considerably from that measured at the 12 foot height and because in very light wind conditions, winds aloft were frequently variable.

A car pace was used exclusively by the pilot in the low airspeed range as a means of speed control for both the upper and lower boundary of the H-V curve. While this system was reasonably effective for the lower boundary, it was considerably less effective along the upper boundary because of the large separation between helicopter and car. It was much more difficult for the pilot to sense relative motion between the two vehicles. Along the lower boundary where the winds were more consistently known, this technique was quite useful. Along the upper boundary, however, indeterminate winds presented problems in approaching a data point (see Pilot's Comments).

Obtaining high hover and near high hover data was again one of the most difficult parts of the test program. Unstable air conditions, indeterminate airspeeds and unknown winds aloft, all contributed to the difficulty. In general, weather conditions prevailing at the test site during the conduct of the project were not as stable as was desired for this type of testing.

The use of the radar altimeter in providing correct information to the pilot for height above the ground was of major importance to the successful completion of the project. The pilot was able to repeat his runs at a constant known height within one or two feet. This made it possible for him to be able to tolerate the lack of precise airspeed information with some degree of confidence.

The wheel landing gear configuration of the test helicopter exposed it to far greater potential damage as a result of a landing gear failure than did the skid gear types of References 1 and 2. As a consequence, the pilot exercised extreme caution during the project as evidenced by the number of runs required to produce data points - a ratio of approximately 8 to 1 - during which time he was constantly evaluating and improving his technique.

Height-Velocity Diagrams

Height-velocity diagrams were first constructed from the experimentally obtained data points. Various cross plots of velocity, altitude, weight, and height-above-the-ground were then constructed and studied to determine what kind of relationships, if any, existed between the many H-V diagrams. Information from these cross plots was then used to adjust the original fairings of the height-velocity curves so that the reworked curves thus obtained provided the best fit with the data points and the cross plotted points. These adjusted curves with the experimental data points are shown in Figs. 4, 5, and 6. The variation with altitude and gross weight for both no-delay and one-second delay conditions is shown in Fig. 7. The variation with altitude for each of the three gross weights tested is shown in Fig. 8. The variation with gross weight for the density altitudes tested is shown in Fig. 9. Since the helicopter tested had limited performance capability at the higher gross weights at the higher altitudes, it was not possible to obtain data over a full range of altitudes at the higher disk loadings. The results herein presented, however, exhibit linear relationships which are quite similar to those obtained from the testing reported in References 1 and 2, and there were no indications that this linear relationship would not hold true for the higher disk loadings at the altitudes tested.

Since the density altitude spread for all the runs at any given test site was larger than desired, an average density altitude for each condition of weight and collective pitch application was derived and utilized to facilitate data analysis. Test points could not be qualified with respect to their relative position about an H-V curve in accordance with their test density altitude alone; i.e., outside the curve for higher altitude and inside the curve for lower altitude because other variables which had much greater effect on the data overshadowed the altitude variation effects.

All of the data points were analyzed on an individual basis as well as from an overall basis to establish their relative position with respect to the H-V diagrams that were developed. The data was generally good for this type of testing and it fit the H-V diagrams very well. In the general analysis there were two basic areas which disclosed data points that fell outside of the developed H-V diagram. These two areas were at Bishop at the 9100 pounds test weight (see Fig. 4 - center curve), and at Lake Tahoe (see Fig. 4 - right hand curve) at the 9100 pounds test weight. Most of these data points occurred along the upper boundary of the curve and were the most difficult to obtain because of conditions which are described in the "Discussion of Tests". The Bishop runs were the initial tests performed in the program and while some points were qualified on an individual basis, it is believed that basic skills had not yet been achieved. The runs at Tahoe are explained on the basis of the pilot's recorded run-by-run comments. Most of the points along the

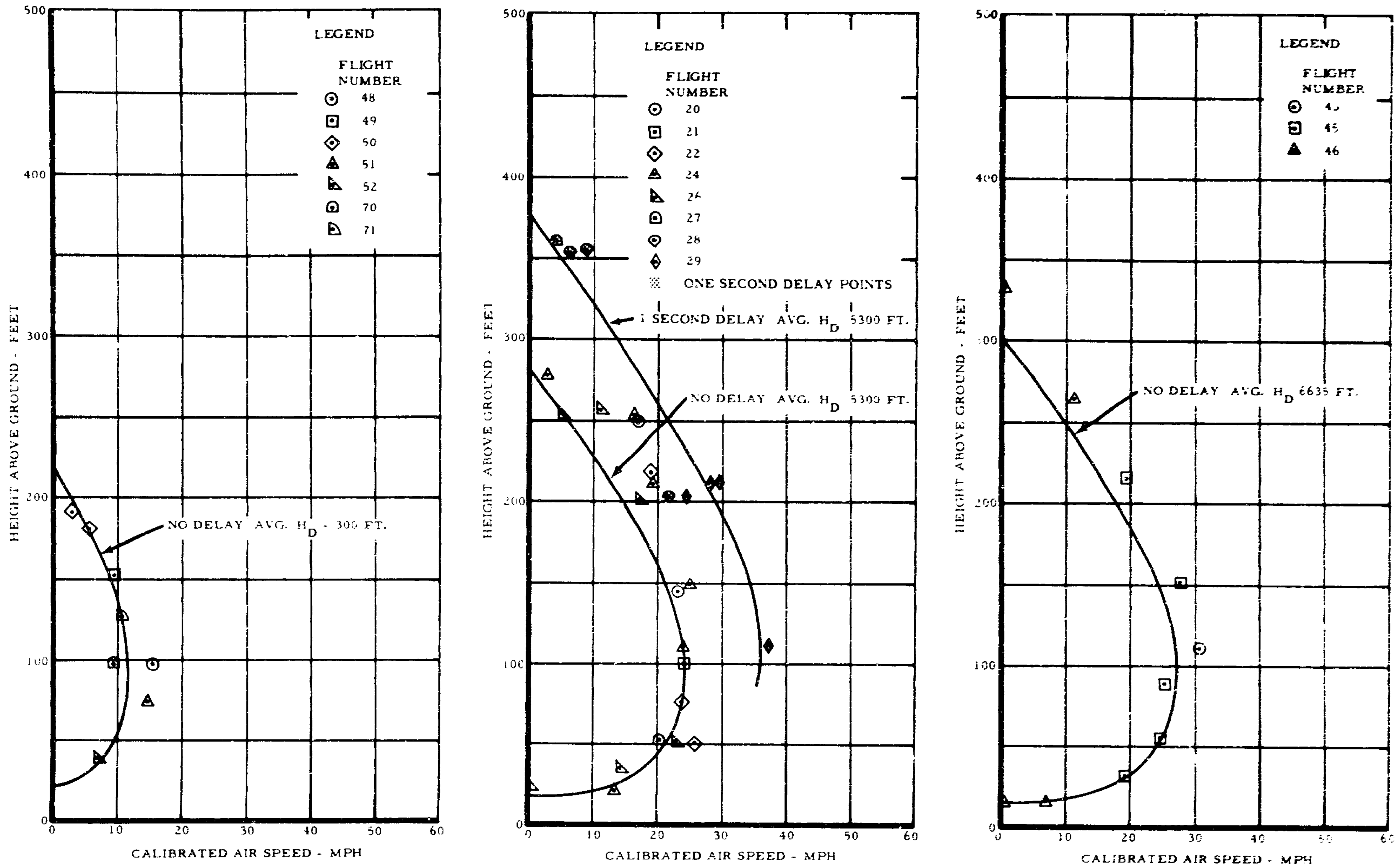

FIG. 4 HEIGHT-VELOCITY DIAGRAMS - BASIC DATA. HELICOPTER GROSS WEIGHT, 9100 POUNDS. THREE DENSITY ALTITUDES SHOWN

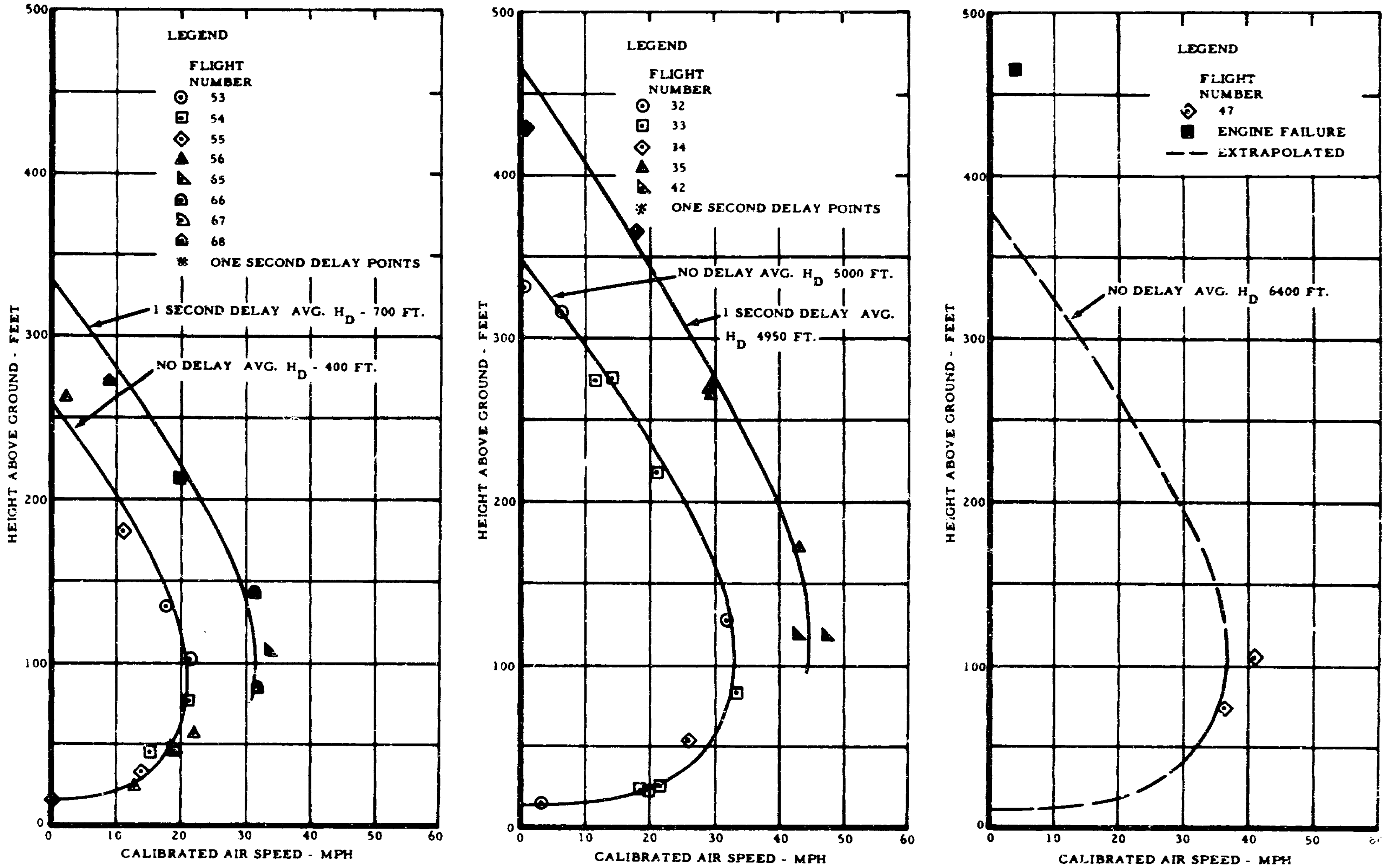

FIG. 5 HEIGHT VELOCITY DIAGRAMS - BASIC DATA. HELICOPTER GROSS WEIGHT, 10,100 POUNDS. THREE DENSITY ALTITUDES SHOWN.

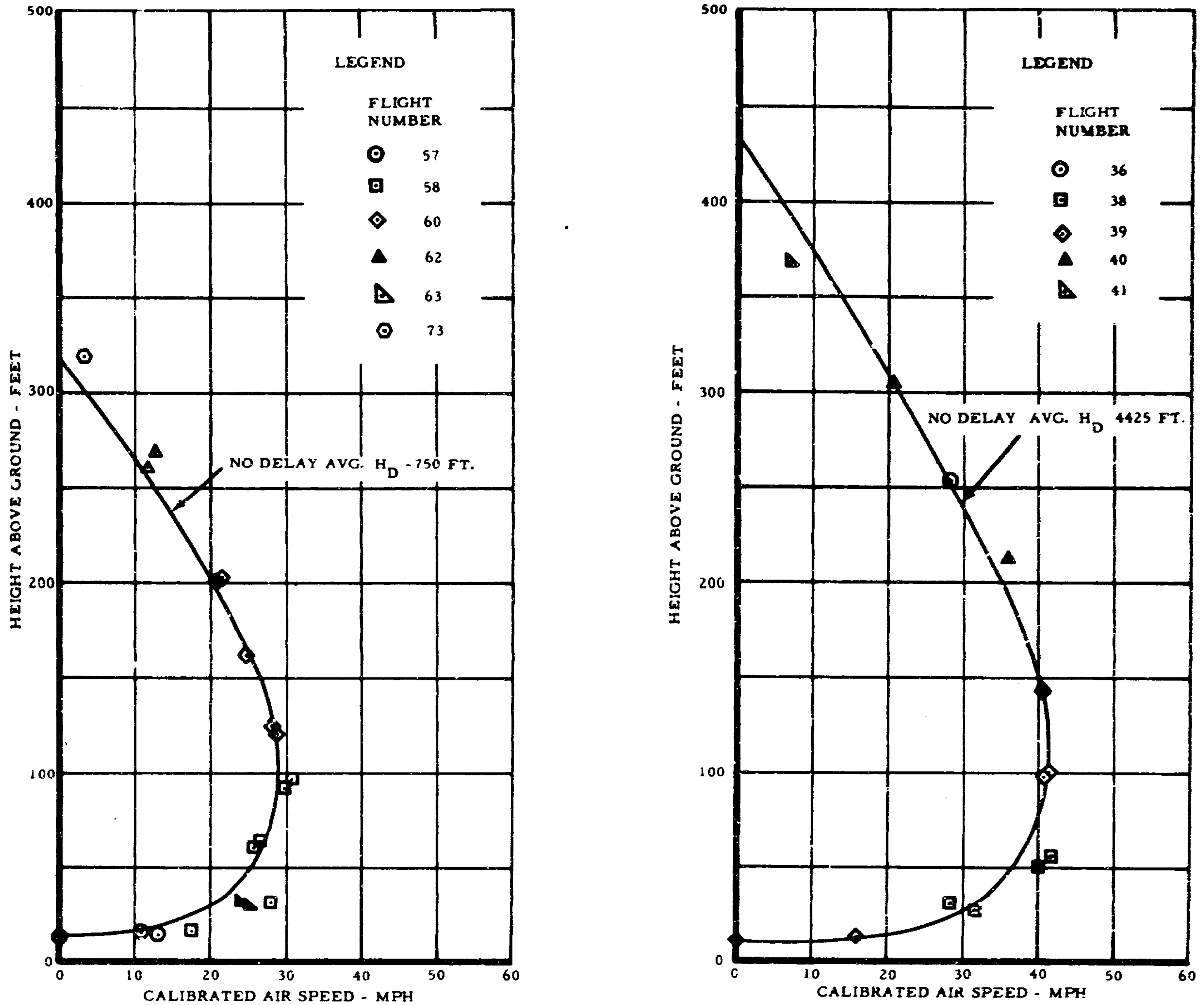

FIG. 6 HEIGHT-VELOCITY DIAGRAMS - BASIC DATA.
HELICOPTER GROSS WEIGHT, 11,100 POUNDS.
TWO DENSITY ALTITUDES SHOWN

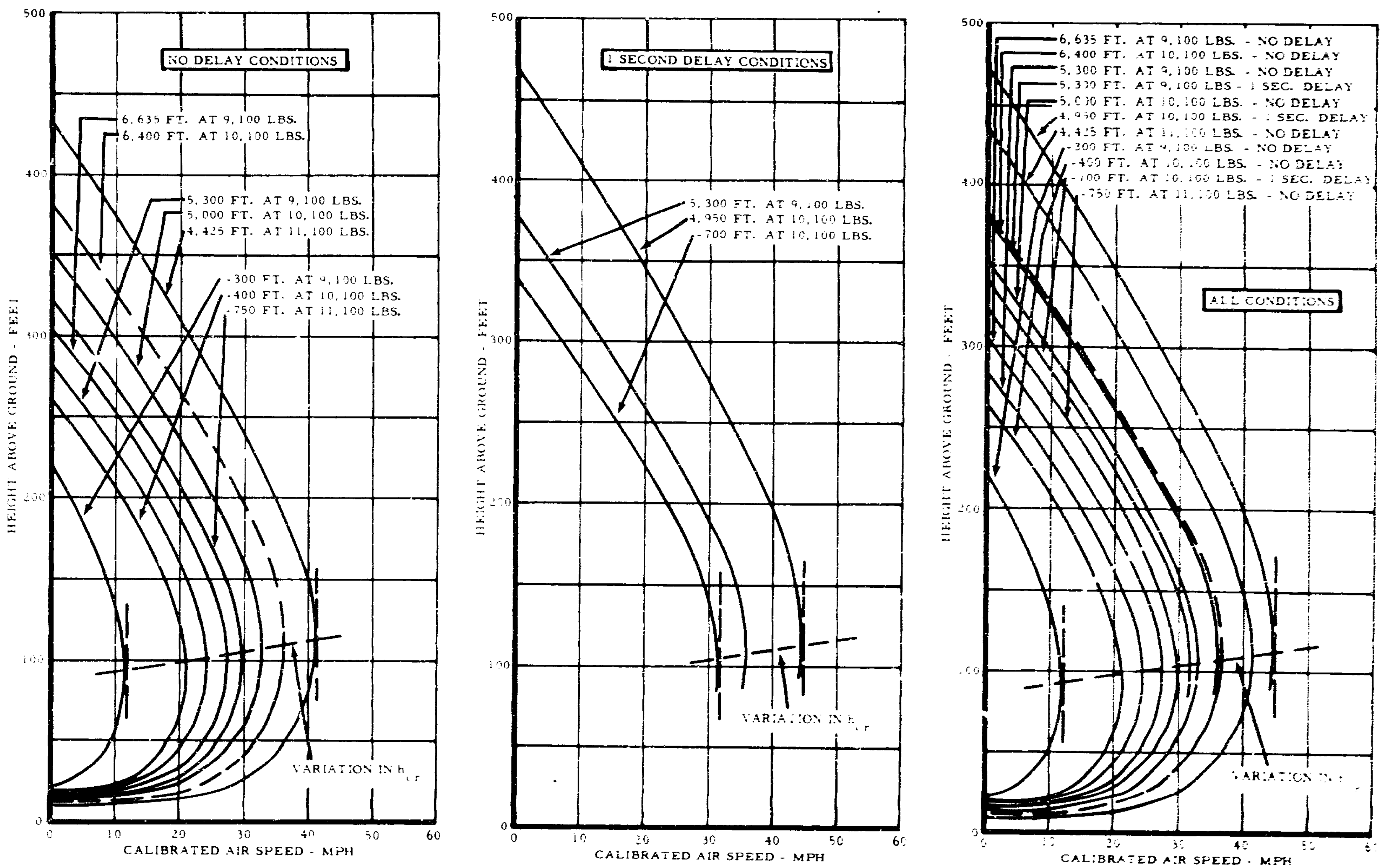

FIG. 7 HEIGHT-VELOCITY DIAGRAM VARIATION WITH DENSITY ALTITUDE AND GROSS WEIGHT DELAY, NO DELAY, AND ALL CONDITIONS SHOWN

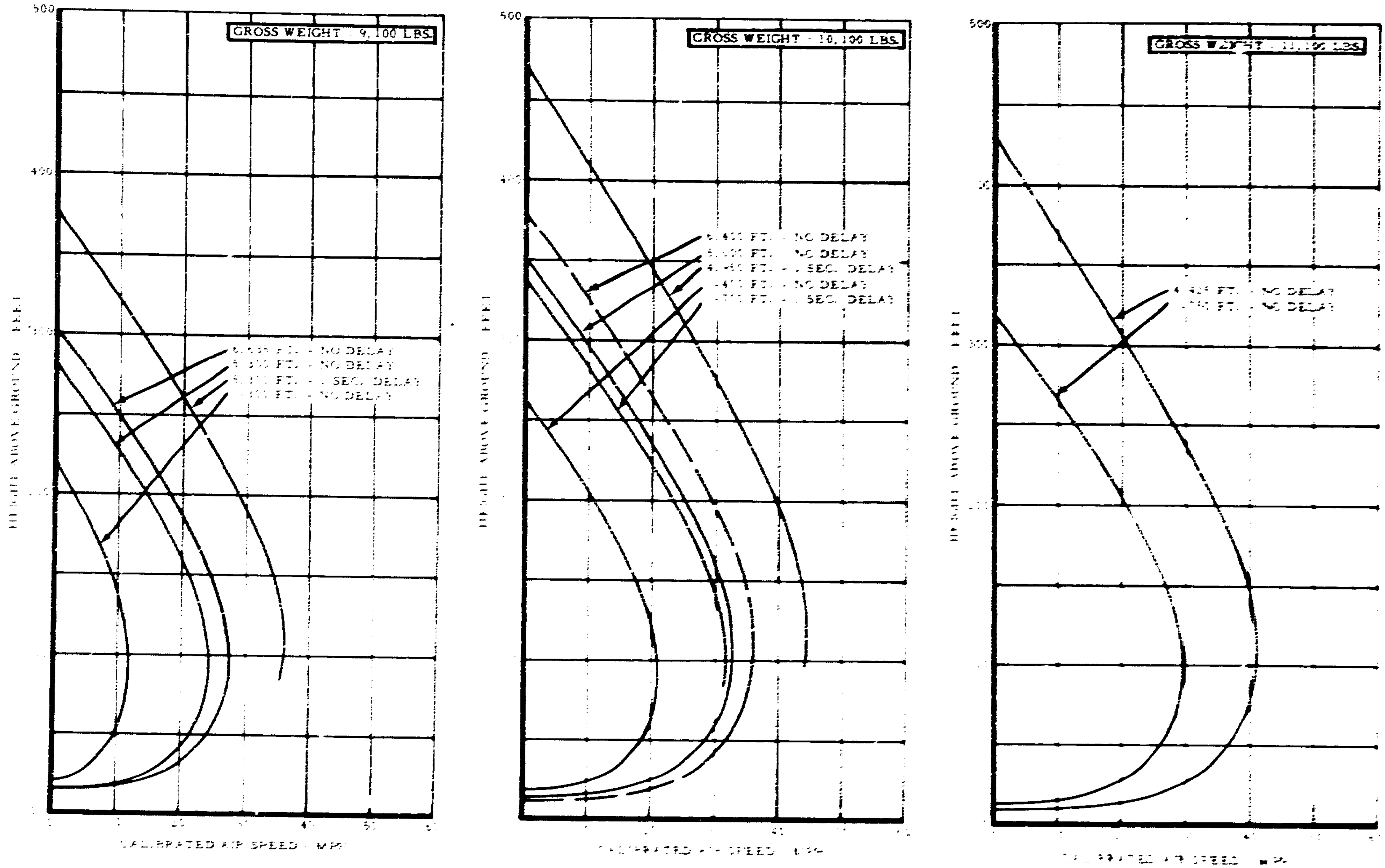

FIG. 8 HEIGHT-VELOCITY DIAGRAM VARIATION WITH DENSITY ALTITUDE GROSS WEIGHTS OF 9100, 10,100, AND 11,100 POUNDS. DELAY AND NO-DELAY CONDITIONS SHOWN

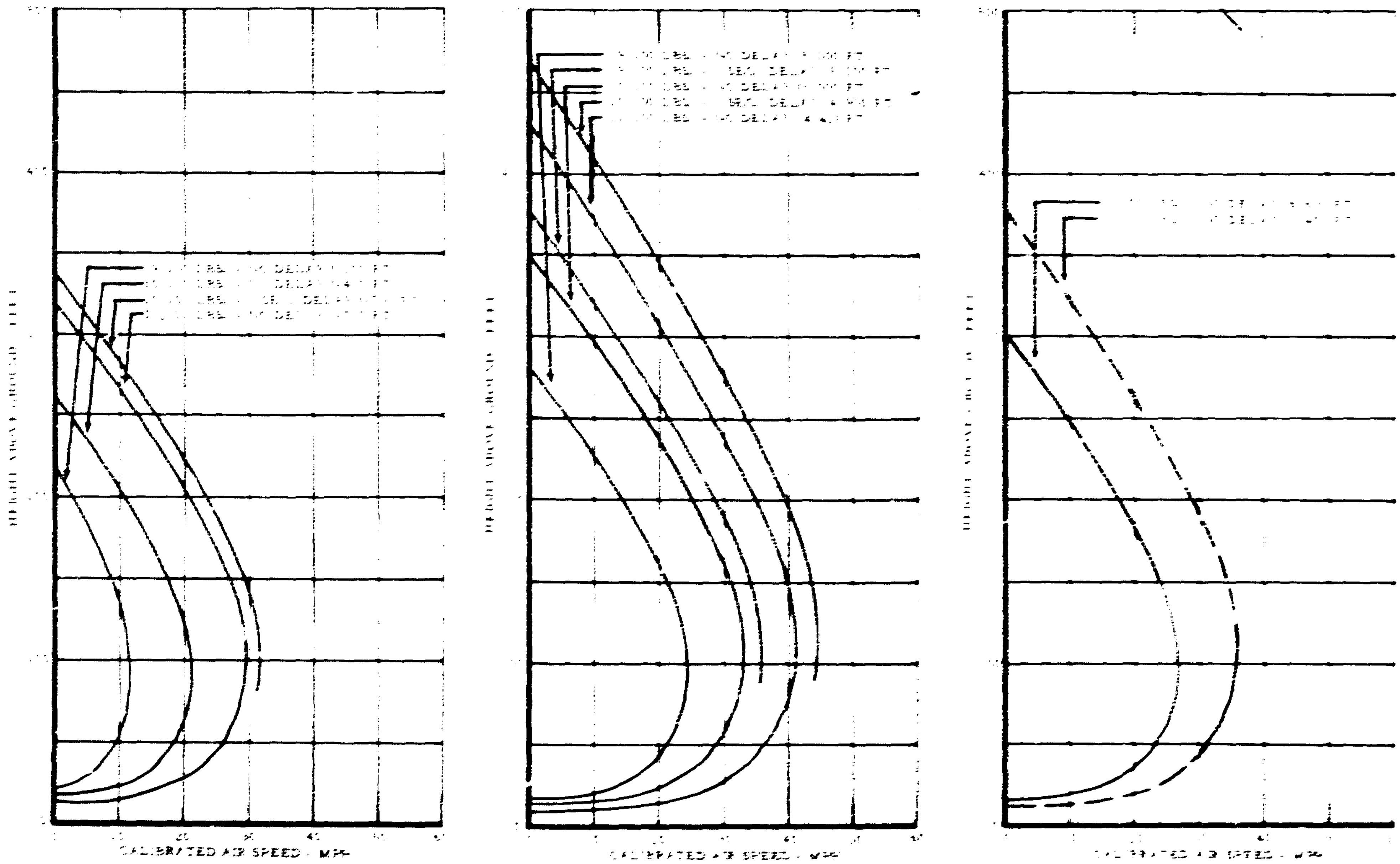

FIG. 9 HEIGHT-VELOCITY DIAGRAM VARIATION WITH GROSS WEIGHT. THREE DENSITY ALTITUDES SHOWN FOR DELAY AND NO-DELAY CONDITIONS

upper boundary were not specifically designated as "solid" points by the pilot, but rather they were designated for investigation with the possibility of better qualifying the point at a later date. Weather conditions and time prevented further testing in this area at Tahoe. Points along the lower boundary from the "knee" down, on the other hand, were designated by the pilot as "solid" points.

All runs which were noted as data points or near-data points by the pilot have been included on Figs. 4, 5, and 6. Examination of the data, however, reveals that in certain areas the pilot was able to reduce the entry speed or change the height appropriately when confirming a particular point. In general, the landing load factors for points lying outside the curve were low and increased as the entry speed was reduced. In a few instances, however, it was noted that lower load factors were obtained when the entry speed was reduced. This was probably due to exceptional pilot technique in executing the maneuver which undoubtedly is the most important single factor in obtaining a maximum performance data point. For the most part, individual points were qualified, concerning their position relative to the faired height-velocity diagrams, on the basis of pilot's comments, landing load factors, density altitude, and the time history analysis.

Table I is a summary chart of the pertinent facts taken from the time histories relative to all of the high hover and near high hover data points. In most cases of high hover or near high hover, stabilizing of the autorotative descent was instituted within 50 feet of descent following throttle chop. That is to say, aft longitudinal stick was applied so that the aircraft started to arrest its nose-down attitude, and in a very gradual manner this was continued to the maximum nose-up attitude (peak of the flare) which occurred at random times prior to touchdown ranging from a quarter of a second to five seconds. In general, where the elapsed time from maximum nose-up attitude to touchdown was short, the landing load factor was on the high side. When the time from maximum nose-up to touchdown was rather prolonged, the landing load factors were relatively low. Correspondingly, the widest variation in landing load factor occurred in the runs from high hover or near high hover. The touchdown speeds (V_{TD}) appear to increase as the weight and altitude increase whether the entry is from high hover or in the "knee" area. The vertical descent velocity following simulated power failure from high hover or near high hover did not show any consistent trends, the highest rates of descent occuring at the 9100 pound gross weight at Bishop. The rates of descent were generally lower, however, for those runs in the vicinity of V_{cr}, h_{cr}, which are listed in Table II. Here again, however, there are no trends with respect to the entry speed, the rates of descent varying as the entry speed increases. With few exceptions, whether entry was from high hover or in the "knee" area, the incremental vertical accelerations following simulated power failure varied between .7 and 1.0 g's.

TABLE I

SUMMARY OF HIGH HOVER (h_{min}) AND NEAR HIGH HOVER DATA

FLT. NO.	RUN NO.	G.W. (1) (LBS)	H_D (2) (FEET)	t_1 (3) (SEC)	A_1 (4) (Δg's)	V_D (5) (FT/SEC)	t_2 (6) (SEC)	h_{TC} (7) (FEET)	t_3 (8) (SEC)	Ω_1 (9) (RPM)	t_4 (10) (SEC)	V_{TC} (11) (MPH)	V_{TD} (12) (MPH)	Ω_2 (13) (RPM)	A_2 (14) (g's)	Ω_3 (15) (RPM)	Ω_4 (16) (RPM)	Ω_5 (17) (RPM)
50	9	9,055	-100	4.69	0.84	-39.0	4.69	180	3.20	208	2.59	5.5	22.7	132	1.48	232		
*50	10	9,125	-100	4.69	0.80	-40.4	4.90	191	1.97	204	0.23	2.8	24.8	147	2.42	234		
55	17	10,045	-200	4.49	0.84	-38.3	4.90	182	3.62	210	2.55	11.0	26.2	127	1.83	235		
*56	4	10,080	-400	5.51	0.74	-43.1	6.12	265	2.03	209	0.88	2.2	31.6	149	2.56	237	194	211
68	9	10,115	-1200	6.53	0.93	-45.1	6.73	274	4.20	201	2.98	8.9	30.8	123	1.49	236	180	208
62	6	11,075	-1300	5.30	0.80	-44.3	6.32	260	4.17	214	2.67	11.9	32.0	147	2.12	234	207	217
62	7	11,050	-1200	5.30	0.93	-45.0	6.12	270	4.41	215	2.57	12.8	29.9	140	1.90	238	207	220
*73	8	11,075	-100	5.71	0.91	-44.8	7.96	320	4.14	211	1.69	3.5	30.8	134	1.55	240	197	223
20	4	9,045	4400	6.66	0.77	-51.0	5.39	249	4.49	210	3.67	16.9	13.8	130	1.66	236	190	214
*24	2	9,100	5200	6.07	0.60	-57.0	4.45	278	2.52	200	1.26	2.6	27.4	156	2.26	237	186	220
24	4	9,045	5300	6.49	0.66	-55.0	5.94	253	4.20	198	3.57	16.2	17.7	134	1.59	238	190	215
26	13	9,075	5600	5.80	0.79	-56.0	4.86	255	2.28	220	1.41	4.7	18.3	145	2.61	238	195	220
26	14	9,060	5500	5.62	0.73	-53.0	4.85	257	3.03	215	2.37	10.8	17.8	138	2.41	237	195	225
*27	9	9,150	5600	7.60	0.82	-54.0	5.95	361	3.24	215	2.49	4.0	19.5	156	1.93	237	180	225
28	6	9,160	5400	7.69	1.07	-54.0	6.20	356	4.68	212	3.86	8.7	14.5	136	2.30	236	172	225
28	7	9,130	5500	8.09	0.88	-53.0	6.29	354	3.60	214	3.07	6.0	14.9	129	2.15	236	170	228
*32	5	10,090	4900	6.86	0.86	-45.8	8.49	332	3.09	222	2.07	0.4	25.6	144	1.39	237	190	222
32	6	10,066	4900	6.94	0.80	-46.0	7.34	316	1.91	227	1.30	6.0	22.9	174	2.47	236	194	230
33	8	10,090	4900	6.08	0.94	-44.5	6.08	276	3.76	221	3.15	14.3	17.1	144	1.71	239	202	228
33	9	10,050	4900	6.26	0.92	-44.9	6.88	275	2.27	222	1.65	11.4	21.2	150	2.56	236	200	222
*34	11	10,065	5050	9.14	0.97	-45.4	7.92	429	5.42	240	5.01	0.7	13.8	131	1.47	238	175	240
34	16	10,125	5200	8.13	0.90	-45.2	6.[illegible]	366	4.95	229	4.34	17.8	19.8	139	1.43	236	182	230
40	9	11,065	4400	6.09	0.89	-44.6	7.[illegible]	305	4.32	226	3.50	20.5	38.7	136	1.44	235	202	233
*41	9	11,085	3650	6.71	0.87	-44.2	9.[illegible]	368	4.76	224	3.74	6.6	35.9	170	1.40	235	200	246
*46	12	9,080	6400	7.12	0.79	-44.9	7.[illegible]	335	4.75	215	3.73	0.1	30.5	130	1.62	239	194	219
46	16	9,050	6300	6.09	0.89	-43.8	7.31	266	4.87	210	2.42	11.3	28.7	129	1.64	237	198	214
**47	11	10,155	6500	8.73	0.71	-44.2	9.34	469	4.60	230	3.78	4.7	45.9	185	1.55	218	205	248

LEGEND

(1) G.W.-TEST GROSS WEIGHT OF THE HELICOPTER
(2) H_D - DENSITY ALTITUDE OF TEST
(3) t_1 - TIME DELAY AFTER THROTTLE OUT BEFORE APPLICATION OF COLLECTIVE PITCH
(4) A_1 - MAXIMUM CHANGE OF ACCELERATION ENCOUNTERED DURING DESCENT
(5) V_D - MAXIMUM RATE OF VERTICAL DESCENT ENCOUNTERED DURING MANEUVER
(6) t_2 - ELAPSED TIME BETWEEN THROTTLE CUT AND ATTAINMENT OF MAXIMUM RATE OF DESCENT
(7) h_{TC} - HEIGHT OVER THE GROUND AT THROTTLE CUT
(8) t_3 - ELAPSED TIME BETWEEN START OF "UP COLLECTIVE" AND TOUCHDOWN
(9) Ω_1 - ROTOR SPEED AT START OF "UP COLLECTIVE"
(10) t_4 - ELAPSED TIME BETWEEN MAXIMUM "NOSE UP" ATTITUDE AND TOUCHDOWN
(11) V_{TC} - CALIBRATED AIRSPEED AT THROTTLE CUT
(12) V_{TD} - CALIBRATED AIRSPEED AT TOUCHDOWN
(13) Ω_2 - ROTOR SPEED AT TOUCHDOWN
(14) A_2 - PEAK ACCELERATION AT TOUCHDOWN
(15) Ω_3 - ROTOR SPEED AT THROTTLE CUT
(16) Ω_4 - MINIMUM ROTOR SPEED OBTAINED AFTER THROTTLE OUT BUT BEFORE TOUCHDOWN
(17) Ω_5 - MAXIMUM ROTOR SPEED OBTAINED AFTER THROTTLE OUT BUT BEFORE TOUCHDOWN

* HIGH HOVER POINTS
** ACTUAL ENGINE FAILURE

TABLE II

SUMMARY OF TYPICAL DATA - AREA OF CRITICAL SPEED (V_{cr}) AND CRITICAL HEIGHT (h_{cr})

FLT. NO.	RUN NO.	G.W. (1) (LBS)	H_D (2) (FEET)	t_1 (3) (SEC)	A_1 (4) (Δ_g's)	V_D (5) (FT/SEC)	t_2 (6) (SEC)	h_{TC} (7) (FEET)	t_3 (8) (SEC)	Ω_1 (9) (RPM)	t_4 (10) (SEC)	V_{TC} (11) (MPH)	V_{TD} (12) (MPH)	Ω_2 (13) (RPM)	A_2 (14) (g's)	Ω_3 (15) (RPM)	Ω_4 (16) (RPM)	Ω_5 (17) (RPM)
48	9	9,080	550	2.84	0.89	-30.7	3.65	97	3.39	208	1.35	15.3	23.8	122	1.49	235		
51	4	9,100	-1000	2.65	1.02	-23.2	3.26	73	2.24	210	0.40	14.6	17.0	145	1.51	236		
70	10	9,155	- 700	2.86	0.77	-26.9	3.47	96	2.38	204	0.98	9.2	15.0	137	2.27	234		
71	5	9,065	- 500	3.67	0.80	-32.4	4.30	126	2.73	205	0.18	10.4	23.1	136	1.69	237		
53	7	10,080	-1350	3.06	1.01	-	-	102	3.02	210	1.90	21.5	27.2	133	1.49	237		
53	13	10,160	- 900	3.47	1.05	-	-	135	2.76	207	1.74	17.8	24.9	133	1.79	236		
54	7	10,065	- 550	2.45	0.80	-23.8	3.26	77	2.99	214	1.77	21.3	24.9	131	1.56	236		
56	9	10,060	- 250	1.84	0.84	-19.4	2.65	57	2.26	217	0.33	22.1	21.2	146	1.96	237		
65	11	10,075	- 400	4.49	0.56	-29.4	5.10	108	1.88	195	0.61	33.6	30.2	141	2.30	238		
66	9	10,065	- 650	3.47	0.68	-24.1	4.08	85	2.41	198	0.78	31.7	31.8	139	1.66	239		
66	12	10,105	- 300	4.69	0.78	-33.4	5.51	144	2.55	196	0.30	31.6	31.4	140	1.76	237	192	202
58	21	11,045	- 500	2.65	0.95	-28.3	3.26	96	2.50	212	1.07	31.[illegible]	31.5	141	1.89	235		
58	22	11,095	- 500	2.86	0.73	-27.5	3.47	92	2.06	215	1.04	30.3	-	148	1.93	234		
58	27	11,050	- 350	1.84	1.10	-20.4	2.65	60	2.34	214	1.72	26.1	26.1	143	2.04	233		
58	28	11,120	- 300	2.04	0.98	-21.0	2.86	63	2.15	220	0.11	27.1	27.1	153	1.78	238		
60	5	11,050	-1650	3.26	0.92	-32.6	3.88	120	2.67	209	0.32	29.1	33.4	146	1.56	238		
60	6	11,040	-1600	3.47	0.92	-31.1	4.08	125	3.12	210	0.47	28.8	32.2	143	1.63	237		
20	11	9,090	4800	4.47	0.58	-40.0	4.14	144	3.87	206	2.65	23.1	17.0	131	1.82	237	195	206
21	9	9,075	5700	3.80	0.84	-32.0	3.68	99	2.73	205	1.71	24.1	14.1	135	1.45	237	199	210
22	14	9,050	5450	3.00	0.66	-31.0	3.20	75	3.68	215	2.46	23.6	16.1	121	1.97	237		
24	11	9,130	6150	3.13	0.98	-35.0	3.31	110	3.06	214	1.78	23.9	13.4	144	1.59	240		
29	7	9,045	5400	4.44	0.45	-36.0	4.11	112	4.39	190	3.37	37.1	19.9	126	1.91	236	190	210
32	10	10,175	5300	3.80	0.78	-32.0	4.22	127	2.14	212	1.31	31.7	24.4	149	1.94	236	205	214
33	12	10,130	5000	3.01	0.90	-24.1	3.62	82	2.70	210	1.68	33.2	17.8	142	1.81	235	214	226
34	4	10,075	4500	2.04	0.96	-16.9	3.06	52	2.46	220	1.44	25.6	14.7	150	1.76	238		
42	4	10,095	3700	3.99	0.80	-29.3	4.62	117	3.11	196	0.23	42.5	35.6	145	1.88	237	192	202
42	5	10,080	3700	4.66	0.72	-26.9	5.30	118	4.17	200	3.12	47.1	40.2	140	1.51	237	193	207
38	11	11,075	4400	1.42	0.71	-15.7	2.82	53	4.43	223	2.83	41.7	29.8	138	1.76	240		
38	21	11,050	4600	2.20	0.70	-21.6	3.02	75	4.16	215	0.69	41.0	29.4	142	1.85	236		
39	3	11,060	4200	3.00	0.60	-27.3	3.82	96	3.97	221	1.42	40.9	37.2	160	1.65	239		
39	4	11,140	4300	2.65	0.85	-26.2	3.88	98	3.40	213	0.44	41.6	33.6	136	2.00	236	212	220
39	11	11,095	4700	3.19	0.90	-34.7	4.21	142	3.81	214	0.85	40.2	36.8	141	1.73	240	212	227
43	8	9,035	6800	3.63	0.88	-26.3	4.24	111	3.29	208	0.64	30.7	29.1	133	1.66	235		
45	15	9,140	6700	3.41	0.79	-23.6	4.02	88	2.71	205	0.67	25.2	26.2	145	1.75	235		
45	19	9,140	6600	2.20	0.72	-16.8	3.00	56	2.71	208	0.69	24.6	19.6	146	1.74	234		
47	4	10,110	6350	3.22	0.66	-28.2	3.63	104	2.92	218	1.49	40.9	38.4	148	1.51	238		
47	9	10,080	6450	2.61	0.76	-22.6	3.43	73	2.27	215	0.43	36.3	37.2	166	1.90	234		

LEGEND

(1) G.W. - TEST GROSS WEIGHT OF THE HELICOPTER
(2) H_D - DENSITY ALTITUDE OF TEST
(3) t_1 - TIME DELAY AFTER THROTTLE CUT BEFORE APPLICATION OF COLLECTIVE PITCH
(4) A_1 - MAXIMUM CHANGE OF ACCELERATION ENCOUNTERED DURING DESCENT
(5) V_D - MAXIMUM RATE OF VERTICAL DESCENT ENCOUNTERED DURING MANEUVER
(6) t_2 - ELAPSED TIME BETWEEN THROTTLE CUT AND ATTAINMENT OF MAXIMUM RATE OF DESCENT
(7) h_{TC} - HEIGHT OVER THE GROUND AT THROTTLE CUT
(8) t_3 - ELAPSED TIME BETWEEN START OF "UP COLLECTIVE" AND TOUCHDOWN
(9) Ω_1 - ROTOR SPEED AT START OF "UP COLLECTIVE"
(10) t_4 - ELAPSED TIME BETWEEN MAXIMUM "NOSE UP" ATTITUDE AND TOUCHDOWN
(11) V_{TC} - CALIBRATED AIRSPEED AT THROTTLE CUT
(12) V_{TD} - CALIBRATED AIRSPEED AT TOUCHDOWN
(13) Ω_2 - ROTOR SPEED AT TOUCHDOWN
(14) A_2 - PEAK ACCELERATION AT TOUCHDOWN
(15) Ω_3 - ROTOR SPEED AT THROTTLE CUT
(16) Ω_4 - MINIMUM ROTOR SPEED OBTAINED AFTER THROTTLE CUT BUT BEFORE TOUCHDOWN
(17) Ω_5 - MAXIMUM ROTOR SPEED OBTAINED AFTER THROTTLE CUT BUT BEFORE TOUCHDOWN

There were two factors which entered into testing which may have been limiting or controlling factors with respect to determining a data point. These factors were blade stall and pitching control. The pilot frequently reported some "shakes" during the flare portion of the landing maneuver. The "shakes" were apparently associated with blade stall. The prevalence of this condition limited his recovery technique at touchdown, thereby controlling his entry speed and height for a given weight and altitude. The other factor was related to the test helicopter's relatively slow response in pitch. It was difficult for the pilot to pitch the helicopter nose-down , pick up airspeed and flare in the time available from entry to touchdown for runs in the close proximity of the "knee". This factor undoubtedly played an important part in the determination of a data point. Both of these factors are discussed in greater detail under "Summary of Pilot's Comments."

Figs. 10 through 12 show a comparison of time history data for high hover, low hover, and the critical speed area for sea level versus high altitude. The figures show that the control inputs and aircraft attitudes are quite similar over the range of altitudes and weights tested. This comparison of the high hover and V_{cr} - h_{cr}, time history data includes all the weights tested to show the effects of weight as well as altitude.

Discussion of One-Second Delay

As a result of the tests of Reference 2, it was learned that a fairly large increase in the size of the height-velocity diagram existed above the knee when a one-second delay was used after throttle cut before collective pitch reduction. It was not known whether this same situation would exist with the high inertia rotor of the S-58 or whether the inertia would cause this displacement to disappear as was indicated by the tests of Reference 1. It was decided, therefore, that the project would be conducted on a no-delay basis with additional testing of one-second delay maneuvers to ascertain what the effect would be. The data obtained utilizing a one-second delay in collective pitch reduction following throttle cut did show a similar displacement as that obtained in Reference 2. Furthermore, the characteristic shape of the one-second delay curve is consistent with the rest of the data defining a family of curves. It would appear, therefore, that there is a specific increase in the size of the H-V diagram above the "knee" as a result of the one-second delay. The tests of Reference 1 were not programmed to seek this out. It should be noted, however, that the one-second delay was only applied above the "knee" in accordance with conventional procedures. The effect of a one-second delay below the "knee" is not known.

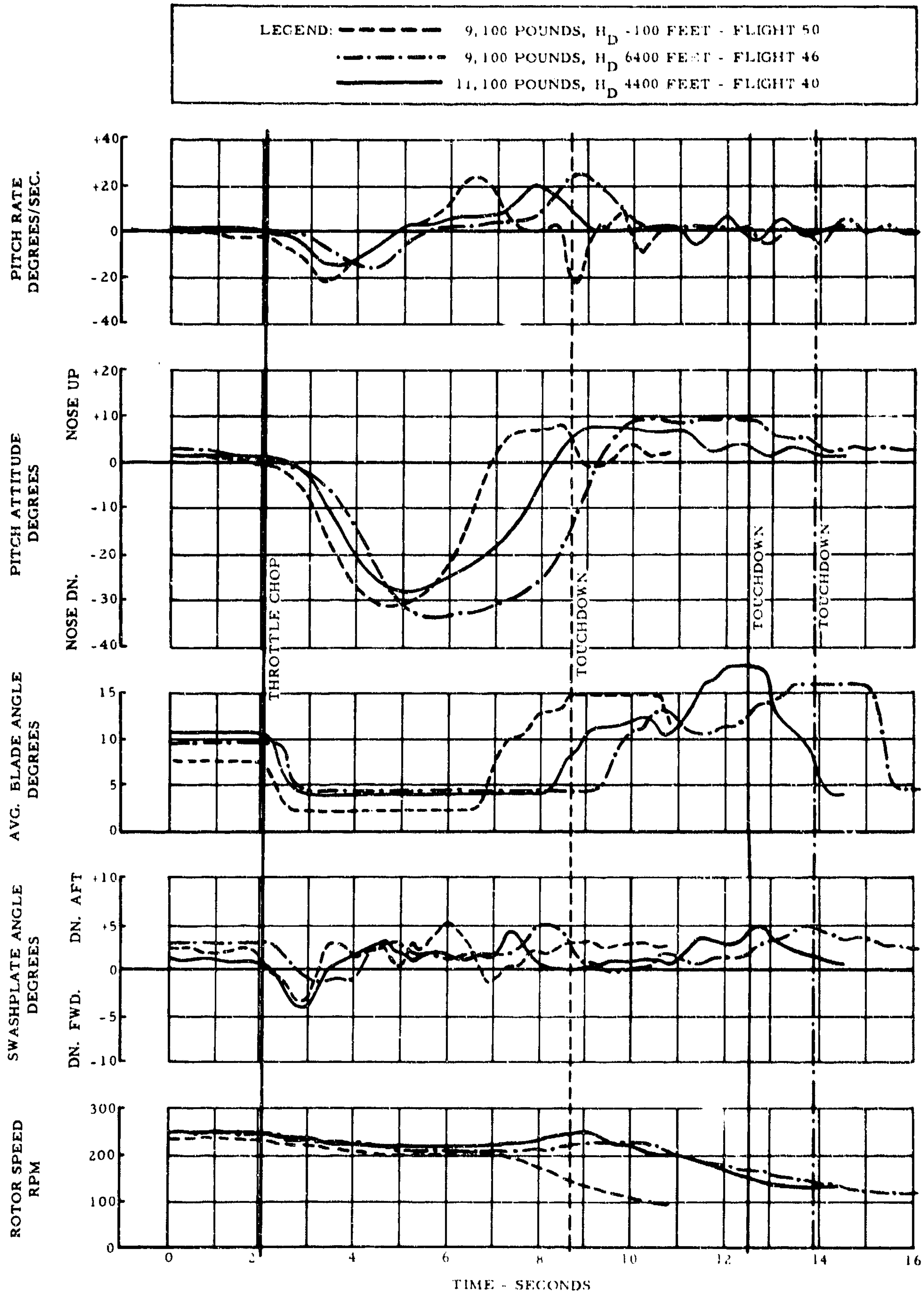

FIG. 10 COMPARISON OF TIME HISTORY DATA FOR HIGH HOVER POINTS

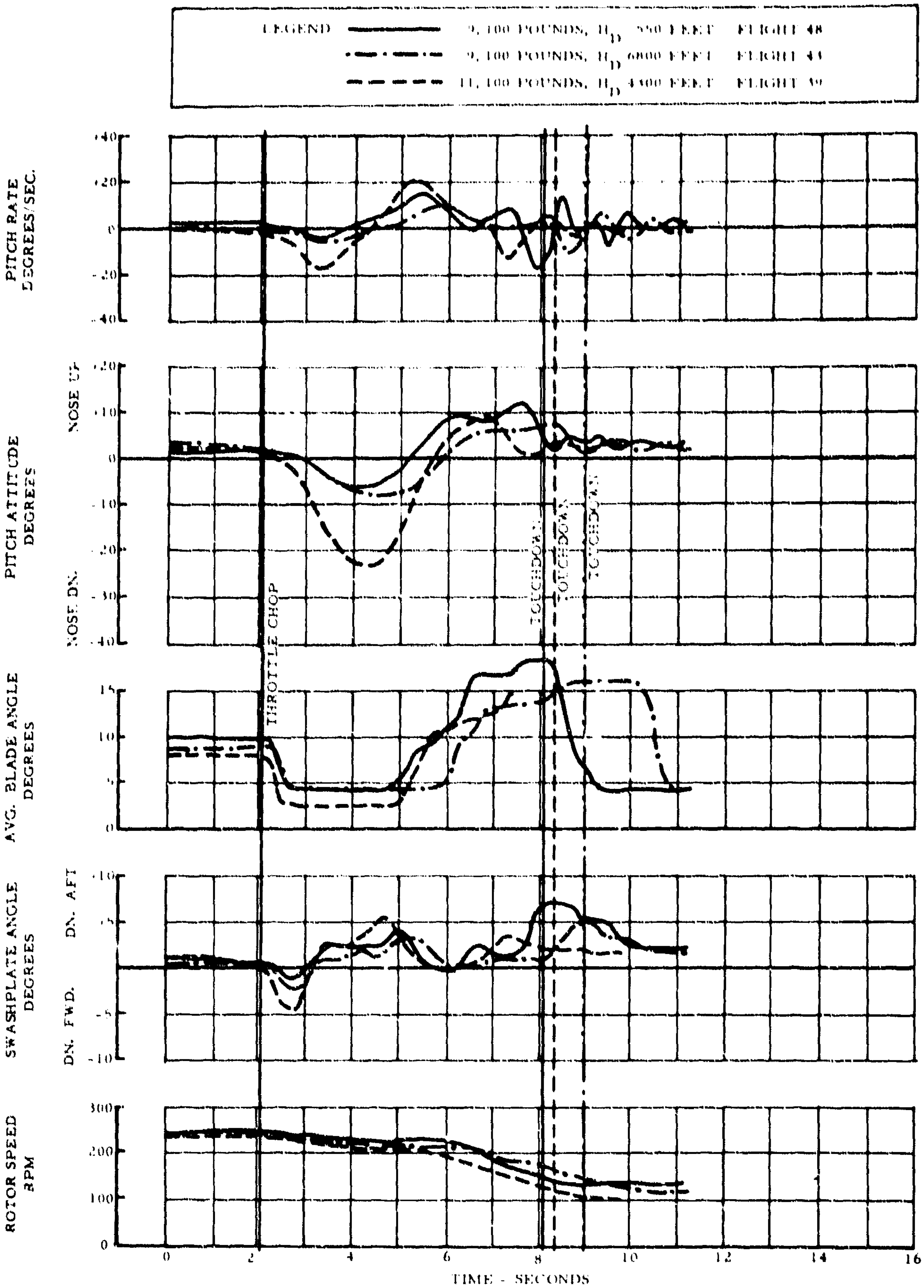

FIG. 11 COMPARISON OF TIME HISTORY DATA FOR CRITICAL SPEED AREA (V_{cr}, h_{cr})

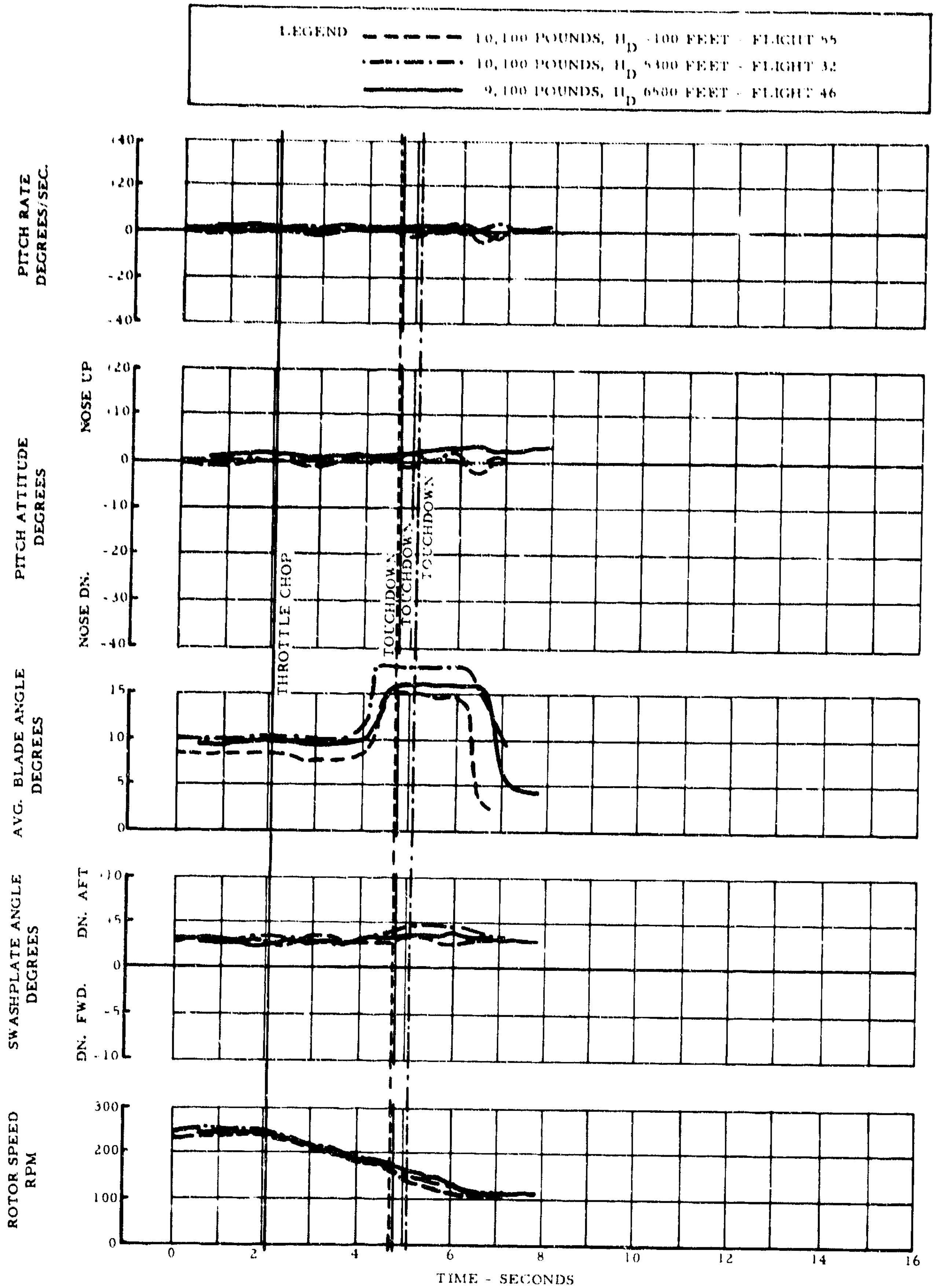

FIG. 12 COMPARISON OF TIME HISTORY DATA FOR LOW HOVER POINTS

Effects of Weight and Altitude

As previously discussed, H-V diagrams were individually drawn through each set of test points and then cross plots constructed of speed versus weight and altitude from which final H-V diagrams were drawn. The controlling points of the H-V diagrams such as V_{cr}, h_{min}, and h_{max} were then cross plotted in a manner to define the H-V diagram relationships.

These cross plots are shown in Figs. 13 through 16. The high hover height, h_{min}, is shown to vary linearly with the square of the critical speed independent of weight, altitude and the time delay in collective pitch reduction as shown in Fig. 17. A set of H-V diagrams resulting from these tests can be partially defined in terms of the critical governing points on the H-V diagram which can be obtained from a set of linear equations. These equations are basically identical to those obtained in References 1 and 2. The differences between these equations and those of the previous tests are in the constants which define the slopes of these linear expressions. The height, h_{cr}, must also be known in order to properly locate the point V_{cr}, h_{cr}. In previous tests, h_{cr} was reported as essentially constant at approximately 95 feet for Reference 1 and as varying between 80 and 100 feet for Reference 2. The current tests clearly indicate that h_{cr} increases with weight and altitude as shown on Fig. 7 by the dotted lines. Throughout the ranges of weights and altitudes tested this height varied from about 90 feet to approximately 110 feet. Inasmuch as the expression shown below for V_{cr} holds true for speeds at heights above and below the height for V_{cr} for approximately 40 to 50 feet as well, the shape of the family of curves is seen to be relativaly constant in the area of the "knee." Therefore, selecting an average h_{cr} of 100 feet would not effect the construction of the H-V diagrams. No attempt was made to establish an expression for h_{cr}.

Equations

1. $V_{cr} = V_{cr}(\text{test}) + C_1 \Delta W + C_2 \Delta H_D$

   where V_{cr} = critical velocity at a given weight and density altitude

   $V_{cr}(\text{test})$ = critical velocity obtained through test

   $$C_1 = \frac{dV_{cr}}{dW}$$

   $$C_2 = \frac{dV_{cr}}{dH_D}$$

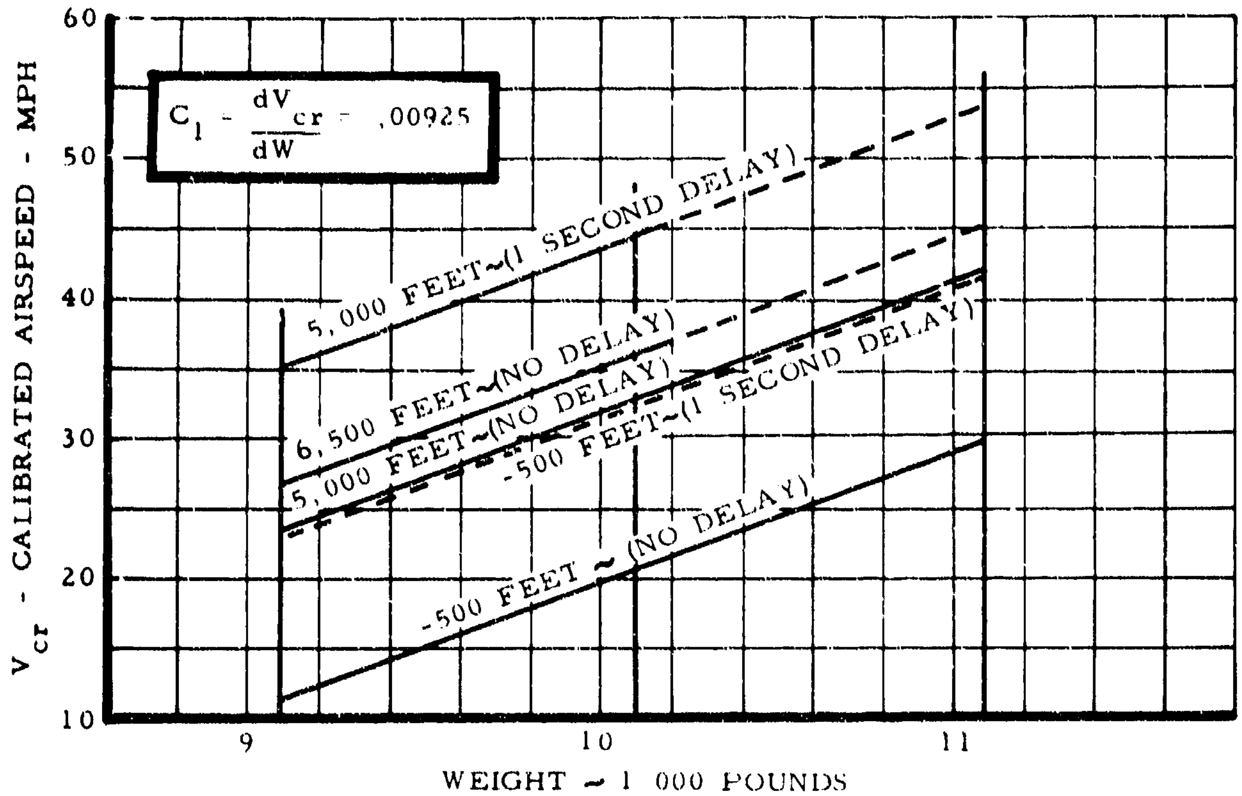

FIG. 13 CRITICAL VELOCITY (V_{cr}) VERSUS AIRCRAFT GROSS WEIGHT FOR THE RANGE OF TEST DENSITY ALTITUDES

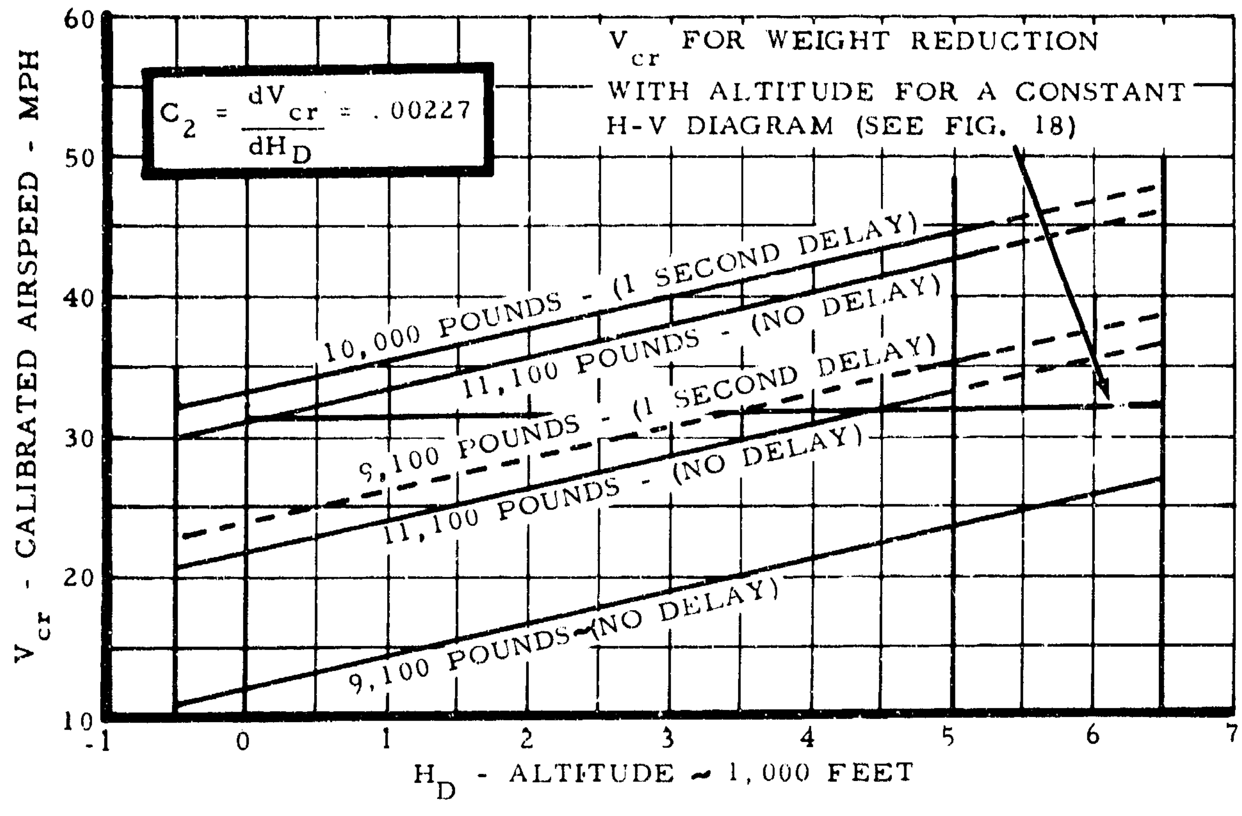

FIG. 14 CRITICAL VELOCITY (V_{cr}) VERSUS TEST ALTITUDE FOR THE RANGE OF TEST WEIGHTS

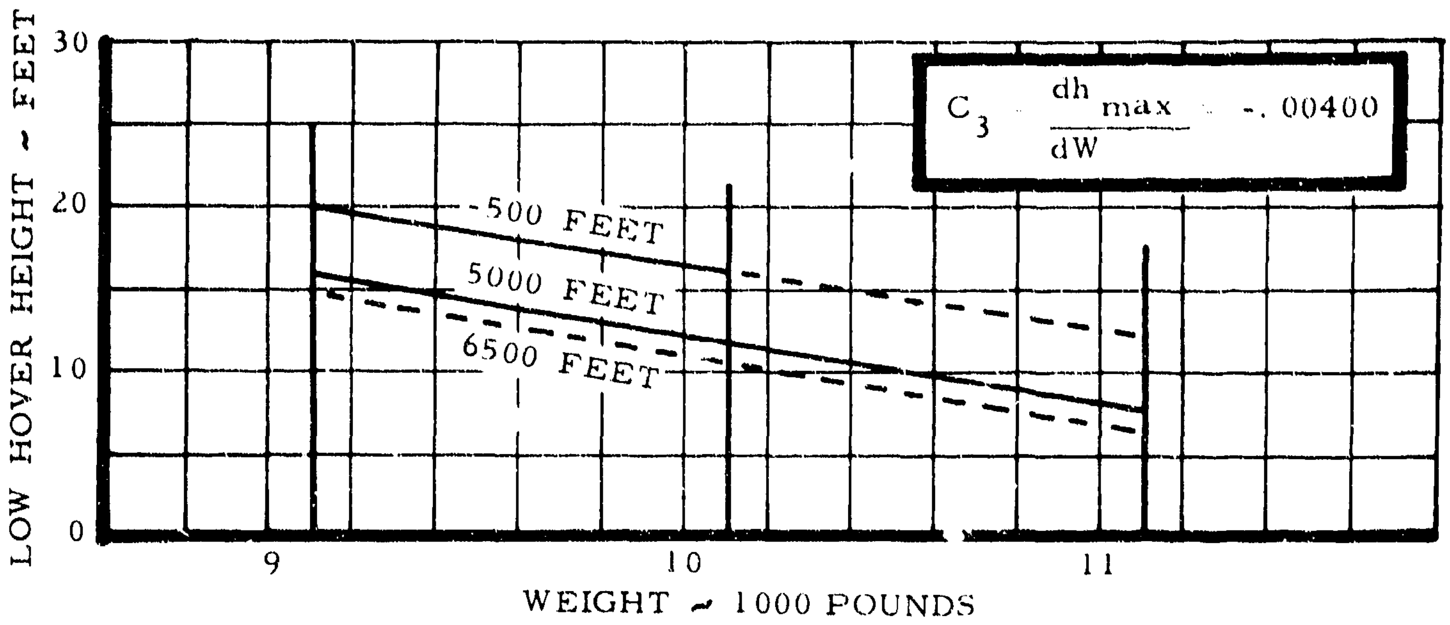

FIG. 15 LOW HOVER HEIGHT (h_{max}) VERSUS AIRCRAFT GROSS WEIGHT FOR THE RANGE OF TEST DENSITY ALTITUDES

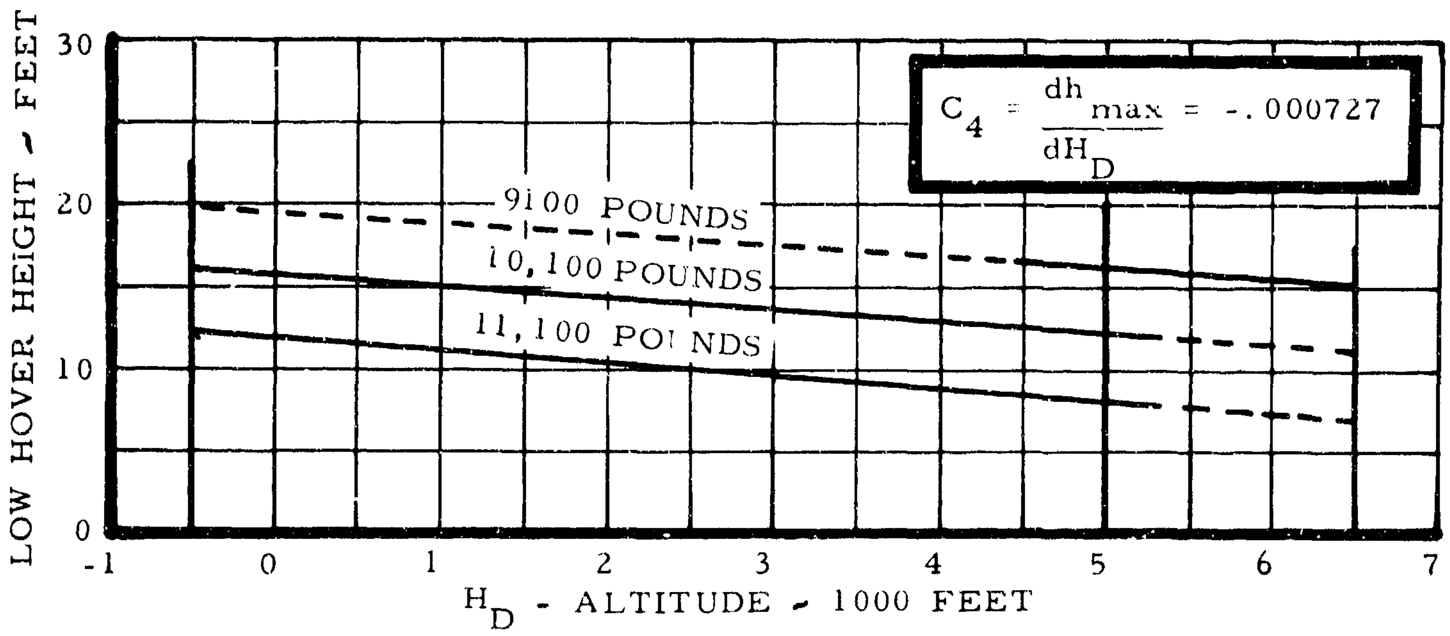

FIG. 16 LOW HOVER HEIGHT (h_{max}) VERSUS TEST ALTITUDE FOR THE RANGE OF TEST WEIGHTS

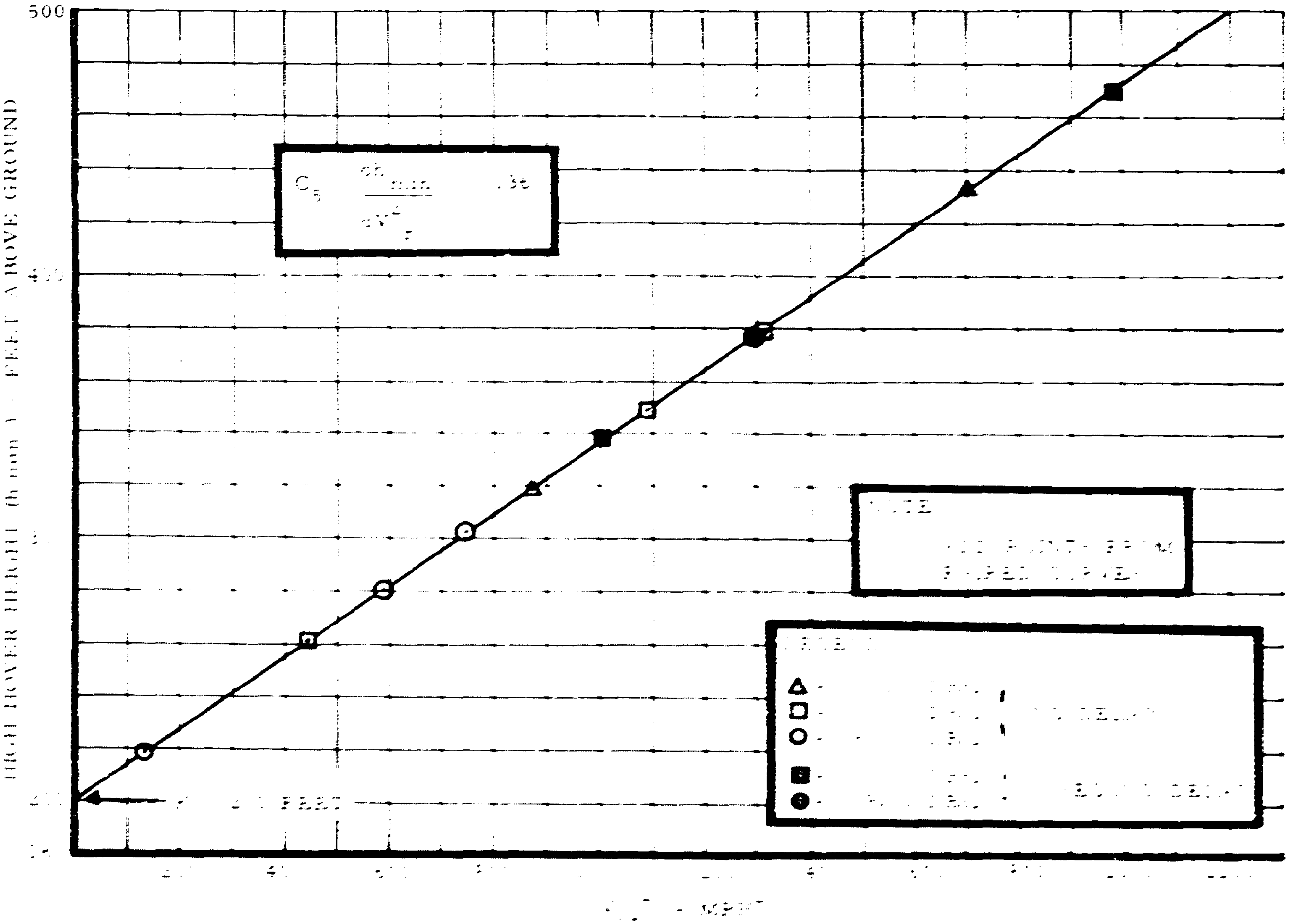

FIG. 17 HIGH HOVER HEIGHT (h_{min}) VERSUS SQUARE OF CRITICAL VELOCITY V^2_{cr}

2. $h_{max} = h_{max}(test) + C_3 \Delta W + C_4 \Delta H_D$

where h_{max} = low hover height at a weight and density altitude

$h_{max}(test)$ = low hover height obtained through testing

$$C_3 = \frac{dh_{max}}{dW}$$

$$C_4 = \frac{dh_{max}}{dH_D}$$

3. $h_{min} = K + C_5 V_{cr}^2$

where K = a constant (the h_{min} intercept)

$$C_5 = \frac{dh_{min}}{dV_{cr}^2}$$

The constants of these empirical equations are applicable only to the test helicopter as were the constants of Reference 1. It is interesting to note, however, that all tests resulted in a set of linear expressions in which only the constants were different. Further, a brief comparative examination of the data of all tests indicates other correlating factors. It appears quite probable, therefore, that a set of equations can be obtained by the application of a nondimensional analysis of the basic parameters and test results of the helicopter used in this project and in the two other projects of this program, which would be applicable to all single engine, single rotor helicopters. Such an analysis might determine whether H-V diagrams can be predicted for a range of weights and altitudes or developed from single weight and altitude test data. No attempt has been made to do this in this report.

Constant H-V Diagram for Reduction of Weight With Altitude

One approach to the problem of establishing an appropriate H-V diagram for variations of weight and altitude is to establish a diagram for maximum gross weight at sea level and hold this diagram constant while reducing weight to compensate for altitude. Such an approach is shown in chart form on Fig. 18. The data for this chart is obtained from Fig. 14 for a constant V_{cr}. Since h_{min} is a function of V_{cr} independent of weight and altitude, the upper part of the diagram is readily obtainable. The lower part of the diagram can be obtained in a like manner from Fig. 16 because the error in h_{max} is negligible.

NOTE:

EXAMPLE SHOWN BELOW IS FOR ILLUSTRATIVE PURPOSES ONLY

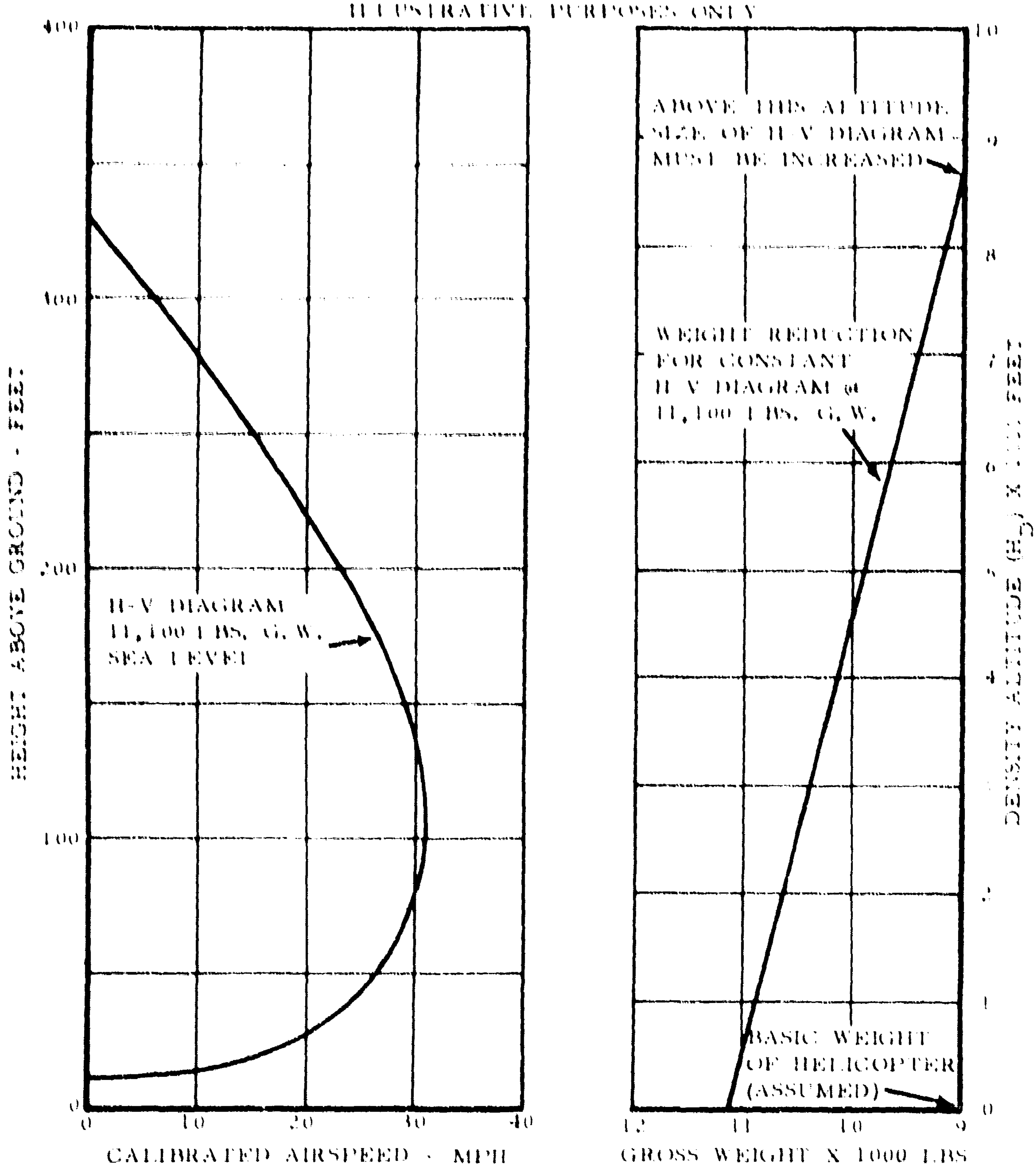

FIG. 18 CONSTANT HEIGHT VELOCITY DIAGRAM WEIGHT REDUCTION

CONCLUSIONS

Based upon the tests of this large single rotor helicopter and an analysis of the test results it is concluded that:

1. The H-V diagrams for this helicopter at different weights and altitudes form a family of curves for the altitudes and weights tested which are defined by a set of equations involving key points on the H-V diagram such as V_{cr}, h_{min}, and h_{max}. These equations show that:

   a. V_{cr} is a linear function of weight or altitude.

   b. h_{max} is a linear function of weight or altitude.

   c. h_{min} is a linear function of V_{cr}^2.

2. The height (h_{cr}) for critical velocity (V_{cr}) increases over the range of weights and altitudes tested varying between 90 to 110 feet. Since the shape of the H-V curves are relatively constant in the area of the "knee," a constant average height of 100 feet for H_{cr} can be assumed without destroying the family relationships of these curves.

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2. Hanley, W. J., and DeVore, G., An Evaluation of the Effects of Altitude on the Height Velocity Diagram of a Lightweight, Low Rotor Inertia, Single Engine Helicopter, Technical Report ADS-46, July 1965.

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3. U. S. Air Force Flight Test Facility, Edwards AFB, California.

4. U. S. NAAF, China Lake, California.

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6. Mr. R. W. Frieburg and the Aircraft Maintenance Base, FAA, Western Region, Los Angeles, California.

7. The many FAA Control Towers, Flight Service Stations, and U. S. Weather Bureau Stations.

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APPENDIX 1

GLOSSARY OF TERMS

(1 page)

APPENDIX 1

GLOSSARY OF TERMS

V_{cr}: critical velocity. The speed above which an autorotative landing can be made from any height after power failure in the low speed regime, mph, CAS.

h_{cr}: the height above the ground at which V_{cr} occurs, feet.

h_{min}: the high hover height - the height above the ground from above which a safe autorotative landing can be made after power failure at zero airspeed, feet.

h_{max}: the low hover height - the height above the ground from below which a safe power off landing can be made after power failure at zero airspeed, feet.

H_D: density altitude at the point of landing, feet.

h: height of the helicopter above the ground, feet.

W: helicopter weight, pounds.

CAS: calibrated airspeed - indicated airspeed corrected for instrument and position error, mph.

APPENDIX 2

TEST AIRCRAFT SPECIFICATIONS AND INSTRUMENTATION DETAILS

(6 pages)

APPENDIX 2

TEST AIRCRAFT SPECIFICATIONS

Significant specifications of the test aircraft are as follows:

1. Powerplant: Wright Model R-1820-84C

    a. Horsepower Ratings

    Takeoff - 1525 hp @ 2800 rpm S. L.
    Maximum Continuous - 1275 @ 2500 rpm S. L.

    b. rpm limitations - 2800 maximum, 2000 minimum

2. Gross Weight:

    a. Maximum certified - 13,000 pounds

3. Hovering Ceiling @ 2700 rpm, standard temperature, 0.00 specific humidity

    a. @ 13,000 pounds - 9000 feet in ground effect

    b. @ 13,000 pounds - 7000 feet out of ground effect

4. Maximum Speed:

    a. Sea level - 107 knots - IAS

5. General Data:

    a. Rotor diameter - 56.0 feet

    b. Rotor disk area - 2460 square feet

    c. Rotor blade chord - 16.4 inches

    d. Blade twist - 4° 40' @ 3/4 blade radius station

    e. Airfoil section - NACA .0012

    f. Number of blades - 4

    g. Solidity ratio - .0569 (.06048 - Eastern Region)
    (.059 Sikorsky)

    h. Disk loading - 5.28 pounds/feet^2 @ 13,000 pounds, maximum G. W.

i. Rotor inertia - 5239 slug $feet^2$

j. Rotor system configuration - fully articulated

k. Flapping hinge offset - 12.0 inches

l. Engine to main rotor ratio - 11.29:1

m. Rotor speed limitations

(1). 258 rpm maximum

(2). 170 rpm minimum

TEST INSTRUMENTATION

A brief description of the test instrumentation utilized for this flight test program is as follows:

1. Airborne - the airborne quantitative information was:

a. Airspeed

b. Altitude

c. Rotor rpm

d. Engine rpm

e. Collective stick position

f. Cyclic stick position

g. Acceleration (vertical, lateral, longitudinal)

h. Fuselage attitude (pitch)

i. Angular velocity (pitch rate)

j. Height (radar altimeter)

k. Instantaneous vertical velocity

l. Throttle position

m. Wheel loads

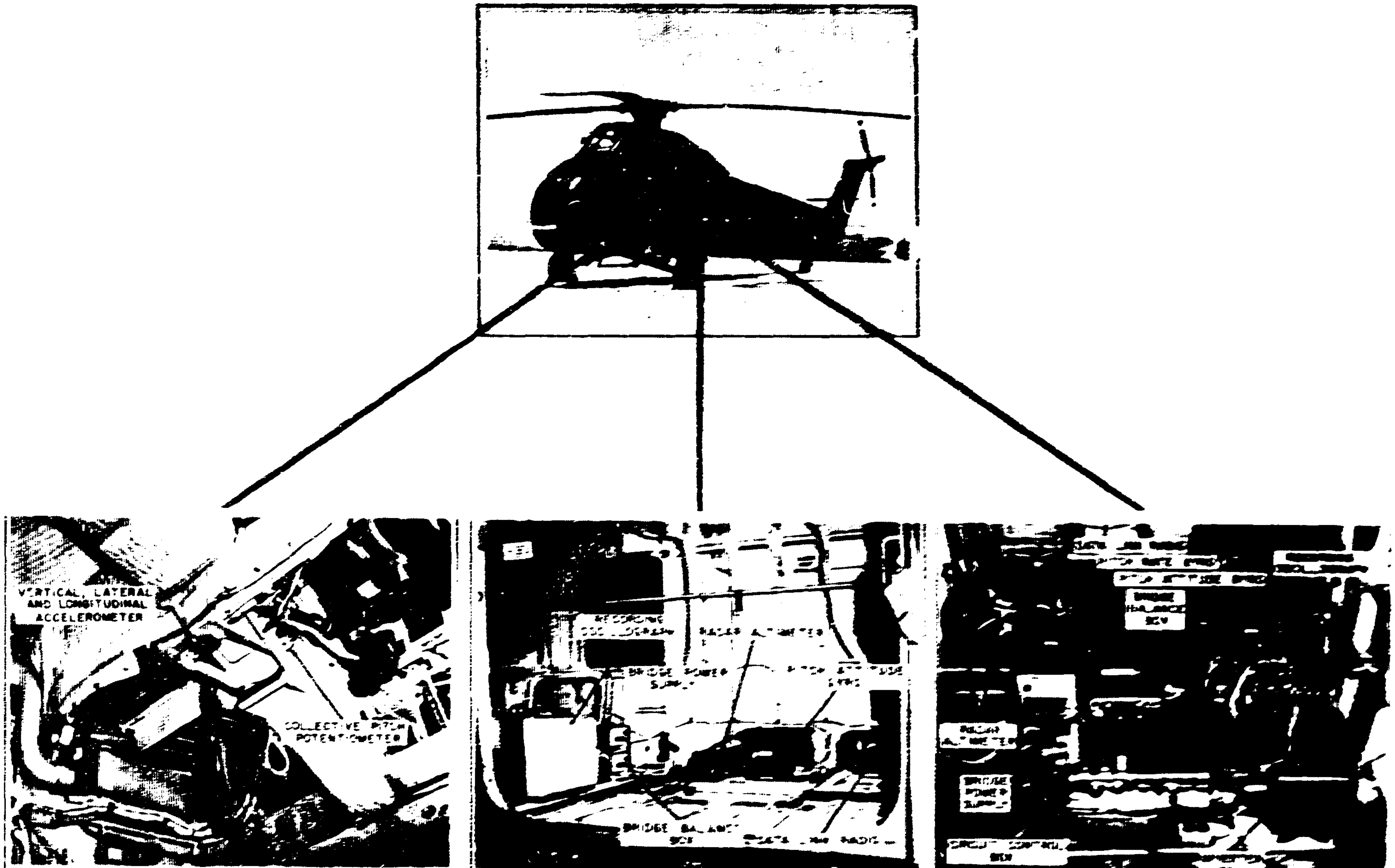
This information was recorded on an oscillograph. Figs. 2-1 through 2-2 show the installation of the recording equipment and some of the basic instrumentation installed on the test aircraft.

2. Ground

Space position equipment utilized for tracking the aircraft is shown in Fig. 2-3.

Correlation of events between the tracking camera and the airborne instrumentation was accomplished by means of a radio data link.

Meteorological equipment utilized for recording atmospheric conditions during the flight tests is shown in Fig. 2-4.



A. VIEW OF FORWARD CABIN - BELOW PILOTS' STATION

B. VIEW LOOKING THROUGH CABIN SIDE DOOR

C. VIEW LOOKING FORWARD - CABIN FLOOR

FIG. 2-1 AIRBORNE INSTRUMENTATION (INTERIOR)

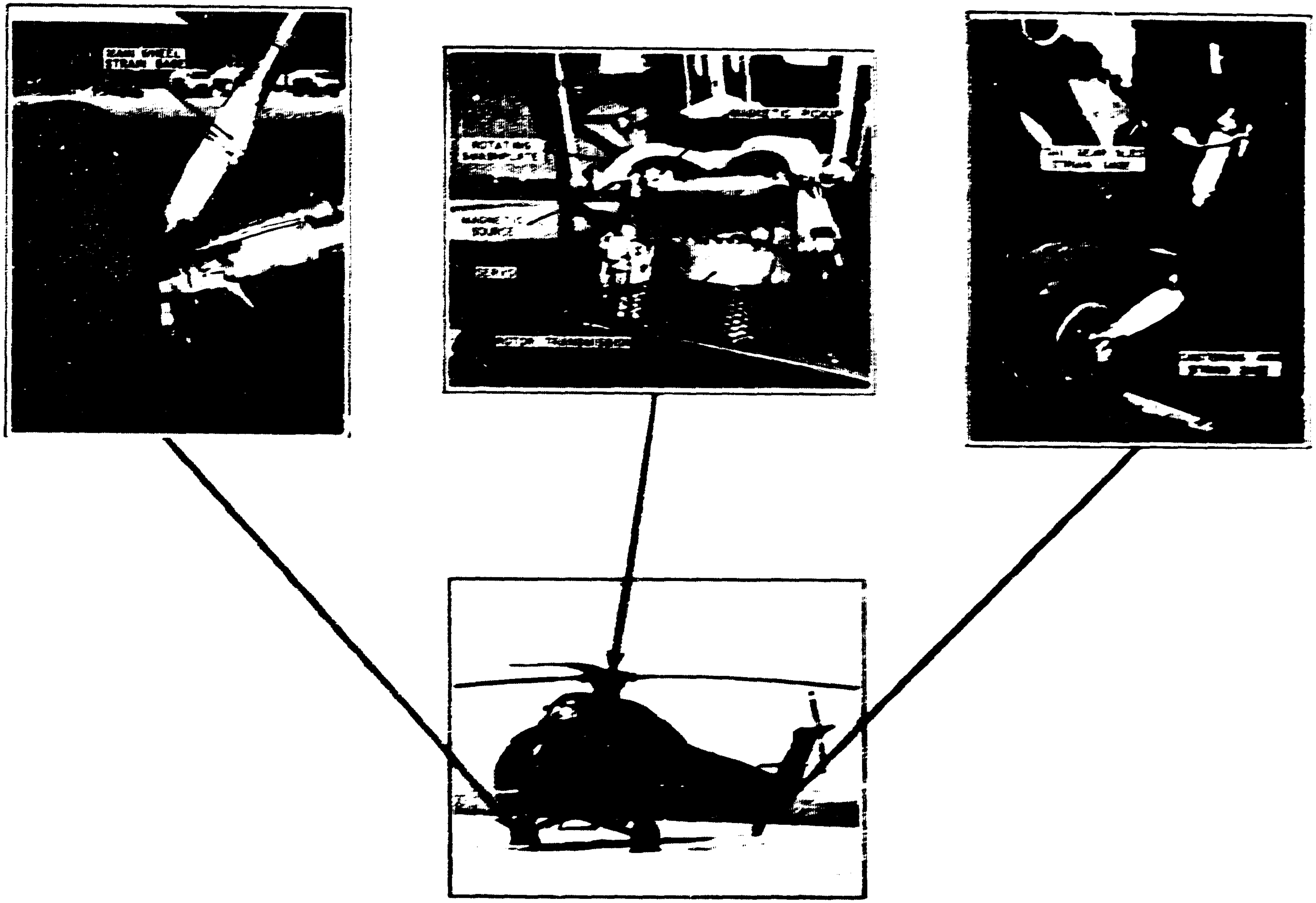


FIG. 2-2 AIRBORNE INSTRUMENTATION (EXTERIOR)

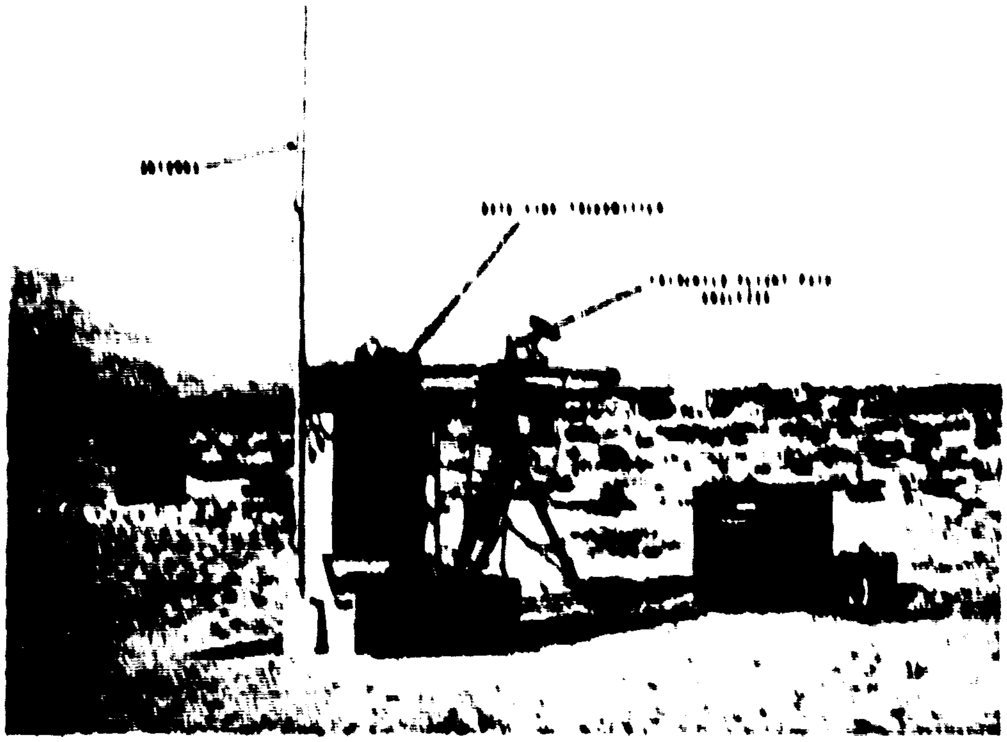


FIG. 2-3 SPACE POSITIONING EQUIPMENT

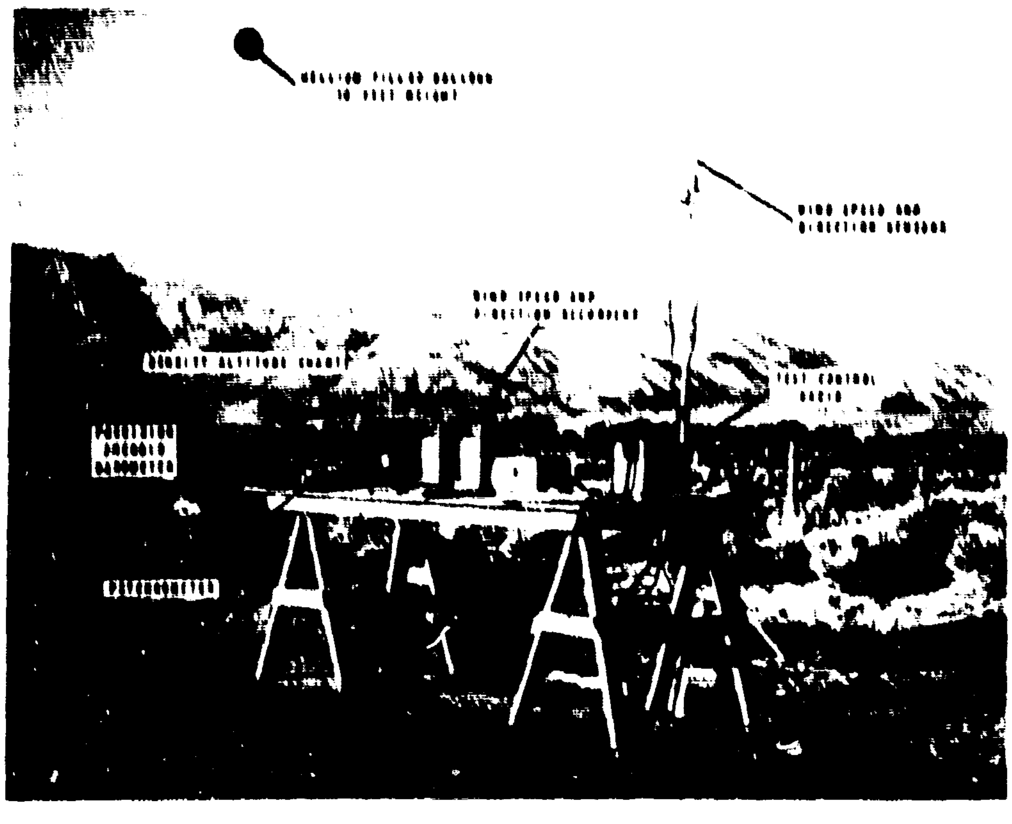


FIG. 2-4 DATA CENTER AND METEOROLOGICAL EQUIPMENT

APPENDIX 3

SUMMARY OF PILOT'S COMMENTS

(7 pages)

APPENDIX 3

SUMMARY OF PILOT'S COMMENTS

PURPOSE

The purpose of these comments is to provide a qualitative analysis of test results and of specific test methods and procedures utilized in the flight tests in order to enable an accurate interpretation of the results to be made. Qualitative comments in this report are based on pilot's comments and notes made in the field and on a qualitative evaluation of the height-velocity characteristics of the test helicopter.

SCOPE

The following areas of the test program will be discussed herein:

a. Pilot techniques and their derivation.

   1. Lower boundary points.

   2. "Knee" and upper boundary points.

   3. One-second delay technique.

b. Data point validation by the pilot.

   c. Entry airspeed control.

   d. Blade stall characteristics during landing.

   e. Stability and control effects on autorotative performance.

   f. Wind and turbulence effects on autorotative performance.

a. Pilot Techniques and Their Derivation

The techniques utilized in this program were developed to accommodate the geometry and handling qualities of the test helicopter and to obtain the maximum autorotative performance inherently available in the vehicle for a given set of entry conditions on a repeatable basis; i.e., the techniques employed were rationalized and applied such that, for a given set of test conditions, use of the same techniques would produce similar test results. This condition of repeatability is inherently essential to an engineering flight test program.

The landing gear structural and geometrical configuration and the overall size of the test helicopter dictated that, for landings utilizing a cyclic flare, the flare should be sequenced to obtain initial touch-down on the tail wheel, utilizing flare energy and partial collective pitch application for this purpose. The remainder of the collective pitch together with aft cyclic control application would then be utilized to cushion the main gear impact. This procedure was designed to take advantage of the energy absorption characteristics of the tail wheel as well as relieve the pilot of the extreme difficulty associated with executing precision autorotative touch-downs in a level attitude (three-point) in this aircraft. This difficulty arises because the pilot's seat is approximately nine to ten feet above ground level thereby increasing the depth perception requirements considerably when gauging the height at which to apply the collective pitch to effect the three-point touch-down. Additionally, the three-point technique requires an extra control motion; i.e., following the landing flare, the helicopter must be re-leveled using cyclic control to affect the three-point touch-down. In large helicopters, this additional control motion is extremely difficult to time accurately due to the relatively low pitch response of the helicopter. That is, the aircraft does not seem to respond readily to pilot control inputs. This apparent lack of response becomes a critical factor when attempting to arrest a high autorotative sink rate by flaring followed by a cyclic control reversal to level the helicopter.

To summarize, the "tail wheel first" touch-down is used in this aircraft because it takes advantage of the energy absorption characteristics of the tail wheel, relieves depth perception requirements and minimizes the effects of lag in helicopter response, thereby enabling the pilot to optimize the precision and repeatability of the landings.

1. Lower Boundary Points

For data points obtained along the lower boundary of the H-V curve, from the low hover point (h_{max}) to an airspeed corresponding to the onset of translational lift (approximately 20-25 mph), the following technique was utilized. Power was manipulated to obtain an airspeed such that relative motion between the aircraft and the pace car was stopped, thereby obtaining the desired throttle chop airspeed. Rotor speed was then adjusted by use of the throttle to obtain 239 rpm $\pm$ 5 rpm. Height above ground was monitored using the radar altimeter to obtain the desired height. The run was then continued in a stabilized condition until entering the test course after which the throttle was closed abruptly. In this segment of the curve close proximity to the ground precluded appreciable reduction of collective pitch following throttle chop. Consequently, the collective pitch control was either held fixed or decreased as possible, following the chop. Little or no cyclic flare was used in this segment of the curve. As the helicopter approached the ground, collective pitch control was applied at a gradually increasing rate such that full control travel was reached just prior to ground impact (± 0.3 seconds). Because no flare was used, ground impact in this area of the curve tended to be in a three point attitude. Following impact, the collective control was reduced to the bottom stop to preclude ground resonance and/or unnecessary aircraft motion, and then the throttle was reapplied to obtain a power-on condition.

2. "Knee" and Upper Boundary Points

For the "knee" and upper boundary areas of the curve, preliminary stabilization was as described above. Following throttle chop, however, the collective pitch control was immediately and firmly reduced to the bottom stop to prevent excessive rotor speed decay and the helicopter was simultaneously nosed over, using forward cyclic control to obtain the nose-down attitude that would produce the desired flare entry airspeed. This attitude was obtained and stabilized as rapidly as possible using large forward cyclic control inputs. Left yawing of the helicopter due to loss of torque was counteracted by using the right anti-torque pedal. As flare height was reached, the helicopter was flared using a rather abrupt aft cyclic control input. As rate of flare reached a peak value, collective control application commenced. Simultaneously, as the helicopter reached the desired flare attitude, the flare rate was terminated and the flare attitude was fixed by application of forward cyclic control. Collective pitch application continued with the objective of reducing tail wheel touch-down sink rates to a low value in order to preclude excessive tail wheel loads from pitching the helicopter nose-down onto the main landing gear. As tail wheel touch-down was obtained, aft cyclic control was applied to utilize remaining flare energy and collective application was continued so that full collective application was obtained just prior to main landing gear impact. Following main gear impact, the collective control was reduced to the bottom stop to preclude ground resonance and unnecessary aircraft motion and throttle was reapplied to obtain a power-on condition.

3. One-Second Delay Technique

One-second delay points were executed essentially as described in Paragraph a.2. above. No unusual difficulty was experienced in executing the throttle chop and the delay prior to collective pitch reduction. A verbal count was employed to time the delay and was initiated when the throttle reached the fully closed position. A conscious effort was made to execute the throttle chop itself within a consistent period of time, usually one to two tenths of a second. Very little helicopter motion was obtained as a result of the throttle chop and no difficulty was experienced in controlling the helicopter. Using a verbal count, it was possible to control the delay time constant within a range of plus or minus two tenths of a second from the desired value and the requirement to verbally count was not distracting to the pilot.

No unusual difficulties were encountered during the descent segment from a delayed throttle chop. There was, however, a marked deterioration in the ability to quickly and accurately obtain the pushover pitching rate which would yield the desired dive angle. This problem was probably due to the decreased pitching control sensitivity at the lower rotor speeds obtained following a delay; i.e., the time required to reach a given dive angle with a given cyclic input increased. It is possible that this factor was accounted for by using larger cyclic inputs but some difficulty was experienced in obtaining the desired response. This characteristic was particularly evident during pushovers from the "knee" of the delay curve since, in this area, due to the limited time available, rapid attitude positioning was critical. On several of the delay points at the "knee", particularly at the heavier weights, it was felt that the slow response in pitching prevented obtaining an attitude which would have yielded better energy utilization.

Significant variations in flare and collective pitch technique were required between the no-delay and delay landings. These variations were introduced primarily to accommodate the possibility of blade stall during the flare and subsequent collective application.

(a) <u>Flare</u>

Flare pitching rates used during delay landings were generally lower than those used for no-delay landings. This was necessary to prevent normal acceleration (g) build-up from causing rotor disk loads which would produce blade stall. This lower pitching rate, although used on almost all delay landings, was probably more apparent when landing from a "knee" throttle chop than when landing from a higher height. This was found to be necessary because "knee" throttle chops tended to produce a decelerating rotor just prior to flare whereas chops from higher heights provided sufficient time for the rotor to begin to accelerate prior to flare. It was observed that the test helicopter's rotor system was particularly susceptible to blade stall when decelerating. Therefore, a lower pitching rate was used on "knee" points.

(b) <u>Flare Height</u>

Because of the lower pitching rates used, it was necessary to provide more time for the helicopter to reach a landing attitude. It is probable, therefore, that flare initiation heights for delay landings were generally higher than those used for no-delay landings. This change in flare height was intuitive on the part of the pilot.

(c) Collective Pitch Application

Collective application, as on no-delay points, was intuitively initiated as rotor speed acceleration peaked (approximately) so as to obtain the maximum benefit from rotor inertial forces. A significantly slower rate of application was used on delay landings, again to preclude the possibility of causing blade stall due to high normal accelerations. Normally, if no stall were obtained, the collective application was made in two segments, the first to cushion the tail wheel touch-down and the second timed to reach maximum collective approximately with main gear touch-down. In a number of landings where blade stall commenced (very noticeable vibration) with the initial application of collective pitch, the rate of application was slowed and then varied to try to prevent complete blade stall while still utilizing all the available energy to land. For landings where blade stall occurred during the flare, full collective application in one pull was required almost immediately since blade stall in the flare produced high sink rates which precluded a gentle application of collective pitch.

b. Data Point Validation By The Pilot

The objectives of this program were to define the height-velocity characteristics of the test helicopter as associated with the attainment of maximum autorotative performance. Obviously, the attainment of maximum performance depends upon the complete utilization of all the energy available to decelerate the helicopter for touch-down, following a throttle chop from a given set of steady-state entry conditions. Assuming that optimum energy utilization is obtained, the landing load factor (g) then becomes a measure of maximum performance. In explanation, if the height-velocity maneuver were initiated from a given set of entry conditions (airspeed and height above ground) and if all the available decelerating energy for landing were utilized, then the attainment of a landing load factor near the design limit for the helicopter under these conditions, would represent a maximum performance point on the height-velocity curve for the density altitude and gross weight being examined. This criteria was employed throughout this test program and produced highly satisfactory results. It is believed that the quantitative measurement of this parameter (landing g loads) as a means of validating the degree of performance of a height-velocity point is, when coupled with pilot qualitative comments, the best approach to obtaining maximum performance data points.

c. Entry Airspeed Control

The use of a pace car to obtain desired entry airspeeds proved to be the most satisfactory means of regulating this parameter in the test helicopter, particularly for data points obtained in the region of the "knee" and along the lower boundary of the curve. Along the upper boundary, however, the accuracy of the pace method deteriorated, primarily due to the distance between the helicopter and the car coupled with the relatively slow speeds required. These factors caused the pilot considerable difficulty in that relative motion between the helicopter and the car could not readily be sensed, thereby causing variations in entry airspeeds. These variations were particularly undesirable when executing a run at an airspeed in the vicinity of a maximum performance point since entry airspeed variations of $\pm$ 2 mph in this area would significantly affect the resulting landing loads.

d. Blade Stall Characteristics During Landing

Blade stall was encountered on several landings in this program. It is also probable that pilot's comments relative to "falling through" the flare in Reference 2 were caused by this phenomena. Because this characteristic, from a maximum energy utilization standpoint, represents landing conditions which are at or in excess of the maximum performance capabilities of the helicopter, its occurrence was a qualitative criteria for determining a maximum performance point. The frequency of occurrence was highest in the area of the "knee" and along the upper boundary of the curve, lowest along the lower boundary of the curve. Because the factors which influence the onset of blade stall do not vary linearly with variations in entry airspeed and height, the probability of its occurrence was extremely difficult to predict. Depending upon the flare entry conditions (airspeed, rotor speed, sink rate) blade stall was encountered either as flare rate reached a peak value and/or as collective pitch application was commenced. Stall was characterized by loss of normal acceleration ("falling through"), random rolling to the left and high amplitude, 4/rev vibration. Normal pilot response was to increase rate of collective application to prevent the impending hard landing. It is probable that the increased rate of application induced additional stalling which consequently resulted in increasing the landing sink rate. It is important to remember, however, that the time available between severe stall onset and ground impact was approximately one to two seconds thereby precluding any reasoned pilot response.

e. Stability and Control Effects on Autorotative Performance

Thus far, two criteria have been discussed as valid means of determining a maximum performance point, normal energy limits and blade stall. A third and last criteria was utilized for several points obtained in this program, that criteria being a stability and control limit. In the test helicopter, particularly in the area of the "knee" of the curve, it was found that insufficient control power was available to pitch the helicopter nose-down following throttle chop and then flare to the desired attitude in the time available between throttle chop and ground contact. Consequently, the pilot was unable to take advantage of all the decelerating energy available in the helicopter.

f. Wind and Turbulence Effects on Autorotative Performance

It is probable that no other variable other than pilot technique has a more pronounced effect on autorotative performance than wind gradients and thermal turbulence. These variables, therefore, if not accurately measured, or accounted for, will induce significant errors in the data which is generated. In this program, throttle chop entry airspeed was gradually reduced during the build-up to a point, by reducing the speed of the pace car a measured amount from run to run. By pacing on the car, the helicopter thus attained a gradually decreasing ground speed. Autorotative performance, however, depends not upon ground speed, but upon true airspeed. If, therefore, during the course of a run, a change in wind speed occurred at a fixed car pace speed, a corresponding change in true airspeed would be obtained, unknown to the pilot. This problem is greatly magnified when a differential wind velocity exists between throttle chop height and the ground, particularly if the wind differential is such that a loss in true airspeed is obtained during the landing sequence (less headwind component on the ground than aloft) since this condition represents an uncontrolled loss of landing energy which is very critical when operating in the vicinity of a maximum performance point.

The same effects are produced by thermal turbulence. If the throttle chop and descent are accomplished with the helicopter located in a rising column of air and if the flight path of the helicopter then carries it outside that rising column prior to landing, it can readily be seen that an uncontrolled, unfavorable loss of landing energy is again obtained.

APPENDIX 4

SUMMARY OF HEIGHT VELOCITY DIAGRAM FLIGHT TEST DATA

(2 pages)

SUMMARY OF HEIGHT - VELOCITY
DIAGRAM FLIGHT TEST DATA

						THROTTLE CHOP		LANDING		
FLT. NO.	RUN NO.	DATE	AIRCRAFT GROSS WT. (LBS)	DENSITY ALTITUDE (FEET)	WIND COMPONENT (MPH)	HEIGHT (FEET)	CAL (MPH)	CAL (MPH)	ACC. (g's)	TIME DELAY (SEC)
20	4	9-21-65	9045	4400	+2.7	249.4	16.9	13.8	1.66	0.01
20	11	9-21-65	9090	4800	+2.0	144.0	23.1	17.0	1.82	0.05
21	9	9-22-65	9075	5700	-1.1	99.4	24.1	14.1	1.45	0
22	4	9-23-65	9040	4500	+1.7	217.6	18.9	11.1	2.10	0
22	14	9-23-65	9050	5450	+2.8	75.1	23.6	16.1	1.97	0
22	17	9-23-65	9155	5750	+2.4	49.7	25.8	12.1	2.10	0
24	2	9-26-65	9100	5200	+1.0	278.0	2.6	27.4	2.26	0
24	4	9-26-65	9045	5300	+1.3	[illegible]	16.2	17.7	1.59	0
24	6	9-26-65	9065	5600	+2.3	211.0	19.1	16.6	2.12	0
24	9	9-26-65	9090	5800	-0.2	149.0	25.0	19.6	1.75	0
24	11	9-26-65	9130	6150	-2.7	109.8	23.9	13.4	1.59	0
24	14	9-26-65	9060	6200	0	19.7	13.2	16.6	1.42	0
26	3	10-3-65	9070	4300	+1.5	33.8	14.2	12.8	2.54	0.05
26	5(2)	10-3-65	9095	4500	-0.7	50.6	22.8	20.5	-	-
26	10(1,3)	10-3-65	9060	5300	-1.3	22.8	0	-	3.38	-
26	13	10-3-65	9075	5600	-1.2	255.0	4.7	18.3	2.61	0
26	14	10-3-65	9060	5500	+2.4	256.9	10.8	17.8	2.41	0
26	16	10-3-65	9050	5750	-1.4	200.6	17.1	22.3	1.93	0
27	2	10-4-65	9075	4800	+1.4	51.3	20.3	12.8	1.81	0
27	9	10-4-65	9150	5600	+1.9	361.0	4.0	19.5	1.93	0.76
28	6	10-5-65	9160	5400	+3.4	356.0	8.7	14.5	2.30	1.17
28	7	10-5-65	9130	[illegible]	+1.5	354.0	6.0	14.9	2.15	1.10
28	11	10-5-65	9100	5750	+3.5	203.0	21.6	22.0	1.56	0.95
29	2	10-5-65	9090	5100	+1.1	204.0	24.6	20.3	2.64	0.99
29	3	10-6-65	9075	5100	+2.2	213.0	29.7	20.7	2.07	1.00
29	4	10-6-65	9045	5200	+0.2	212.8	27.9	17.9	2.13	0.97
29	7	10-6-65	9045	[illegible]	+1.5	112.0	17.1	19.9	1.91	1.09
32	[illegible]	11-6-65	10,090	4900	-2.2	332.0	0.37	25.6	1.39	0.10
32	6	11-6-65	10,066	4900	+1.2	316.0	6.0	22.9	2.47	0.05
32	10	11-6-65	10,175	5300	-1.3	126.5	31.7	24.4	1.94	0
32	14	11-6-65	10,125	5300	+1.1	12.1	2.9	2.4	2.21	0
33	5	11-7-65	10,050	4750	-3.2	218.4	20.9	16.6	2.41	0
33	8	11-7-65	10,090	4900	-1.7	276.0	14.3	17.1	1.71	0
33	9	11-7-65	10,050	4900	-2.3	274.6	11.4	21.2	2.56	0
33	12	11-7-65	10,130	5000	-3.2	81.5	33.2	17.8	1.81	0
33	15	11-7-65	10,075	5100	+3.1	23.4	21.1	19.7	1.87	0
33	16	11-7-65	10,050	5100	+3.4	21.4	12.6	[illegible]	2.01	0
33	17	11-7-65	10,100	5200	+3.2	21.5	18.2	[illegible]	2.15	0
34	4	11-8-65	10,075	4500	0	51.5	25.6	14.7	1.76	0
34	11	11-8-65	10,065	5050	-3.5	429.0	0.74	13.8	1.47	0.96
34	16	11-8-65	10,125	5200	-1.5	366.0	17.8	19.8	1.43	0.96
35	8	11-11-65	10,125	4750	-2.2	267.0	28.5	31.4	1.33	1.02
35	10	11-11-65	10,075	4800	-1.2	263.4	29.0	30.0	1.29	1.10
35	12	11-11-65	10,105	4900	+1.6	270.0	29.4	25.4	1.32	0.82
35	15	11-11-65	10,050	5000	+1.5	171.5	43.0	31.7	1.47	0.82
36	10(3)	11-12-65	11,080	5000	0	253.0	26.1	[illegible]	3.17	0.05
38	5	11-20-65	11,105	4350	+2.6	25.3	31.6	23.8	1.81	0
38	6	11-20-65	11,090	4400	+1.3	28.8	28.3	22.6	2.01	0
38	10	11-20-65	11,100	4400	+0.75	48.5	40.0	30.8	1.87	0
38	11	11-20-65	11,075	4400	+1.8	53.2	41.7	29.0	1.76	0
38	21	11-20-65	11,050	4600	+0.7	75.5	41.0	29.4	2.15	-
39	3	11-21-65	11,060	4200	+2.7	95.6	40.9	37.2	1.65	0
39	4	11-21-65	11,140	4300	+0.7	98.0	41.6	33.6	2.00	0
39	11	11-21-65	11,095	4700	+0.2	142.0	40.2	36.8	1.73	0
39	14	11-21-65	11,060	4650	+2.4	3.6	0	-	2.14	0
39	16	11-21-65	11,110	4800	+1.5	11.1	15.7	13.6	1.99	-
40	4	11-22-65	11,060	4150	+3.0	212.0	35.8	36.8	1.47	0
40	9	11-22-65	11,065	4400	+1.2	304.6	20.5	38.7	1.44	0
41	9	11-26-65	11,085	3650	-1.5	368.0	6.6	35.9	1.40	0.04
42	4	11-26-65	10,095	3700	-1.7	116.8	42.5	35.6	1.88	1.00
42	5	11-26-65	10,080	3700	+0.7	118.0	47.1	40.2	1.51	1.12
43	8	12-2-65	9,095	6800	-1.3	111.0	30.7	29.1	1.66	0.06
45	5	12-5-65	9,105	7100	+2.6	151.0	27.7	26.2	1.56	0
45	11	12-5-65	9,065	6950	-1.1	216.0	19.2	23.2	1.76	0
45	15	12-5-65	9,140	6700	-0.8	88.4	25.2	26.2	1.75	0

SUMMARY OF HEIGHT - VELOCITY
DIAGRAM FLIGHT TEST DATA

PLT. NO.	RUN NO.	DATE	AIRCRAFT GROSS WT. (LBS)	DENSITY ALTITUDE (FEET)	WIND COMPONENT (MPH)	THROTTLE CHOP HEIGHT (FEET)	THROTTLE CHOP V CAL (MPH)	LANDING V CAL (MPH)	ACC. (g's)	TIME DELAY (SEC)
45	19	12-5-65	9140	6000	+0.1	55.8	24.6	19.6	1.74	0
45	22	12-5-65	9070	6400	-0.4	31.0	19.2	16.2	2.21	0
46	3	12-6-65	9120	6500	-1.3	14.9	0	-	2.36	0
46	6	12-6-65	9060	6600	+3.4	15.6	6.8	-	2.07	0
46	12	12-6-65	9080	6400	-2.1	335.0	0.1	30.5	1.62	0
46	16	12-6-65	9090	6300	-1.9	265.8	11.3	28.7	1.64	0
47	4	12-10-65	10,110	6350	+3.8	104.0	40.9	38.4	1.51	0
47	9	12-10-65	10,080	6450	+1.7	73.1	36.3	37.2	1.90	0
47	11(4)	12-10-65	10,155	6500	+1.1	468.8	4.7	45.9	1.55	0
48	9	1-11-66	9080	550	+2.8	96.8	15.3	23.8	1.49	0
49	7	1-14-66	9045	300	-0.8	151.8	9.2	21.8	1.82	0
50	9	1-18-66	9055	-100	-2.9	180.0	5.5	22.7	1.48	0
50	10	1-18-66	9125	-100	-1.6	191.0	2.8	24.8	2.42	0
51	4	1-19-66	9100	-1000	-2.6	73.0	14.6	17.0	1.31	0
52	8	1-21-66	9115	- 700	-1.4	38.3	6.7	6.3	1.97	0
53	7	1-22-66	10,080	-1350	+2.8	102.0	21.5	27.2	1.49	0
53	13	1-22-66	10,160	- 900	+3.2	135.0	17.8	24.9	1.79	0
54	7	1-23-66	10,065	- 550	+4.9	77.3	21.3	24.9	1.56	0
54	12(2)	1-23-66	10,100	- 400	0	45.0	15.1	15.0	-	0
55	7	1-24-66	10,080	- 550	+2.6	33.0	13.9	13.9	2.56	0
55	17	1-24-66	10,045	- 200	0	181.5	11.0	26.2	1.83	0
55	20(1)	1-24-66	10,080	- 100	+2.3	15.8	0	-	2.13	0
56	4	1-25-66	10,080	- 400	0	205.0	2.2	31.6	2.56	0
56	9	1-25-66	10,060	- 250	+1.6	57.0	22.1	21.2	1.96	0
56	11	1-25-66	10,095	- 100	-2.3	24.8	12.8	10.7	2.62	0
57	4(1)	1-26-66	11,125	- 500	+3.8	11.2	0	-	2.32	0
57	7	1-26-66	11,050	- 400	+2.3	14.4	11.4	11.0	2.75	0
57	11	1-26-66	11,110	- 150	0	13.0	13.6	11.5	2.29	0
58	5	1-27-66	11,075	- 750	-1.5	15.0	18.0	18.0	2.11	0
58	16	1-27-66	11,100	- 550	+4.0	30.8	28.7	28.7	1.65	0
58	21	1-27-66	11,045	- 500	+1.5	76.0	31.3	31.5	1.89	0
58	22	1-27-66	11,095	- 500	0	91.3	30.3	-	1.93	0
58	27	1-27-66	11,050	- 350	-3.0	60.0	26.1	26.1	2.04	0
58	28	1-27-66	11,120	- 300	-1.6	63.0	27.1	27.1	1.78	0
60	5	1-29-66	11,050	-1650	+2.1	120.0	27.1	33.4	1.56	0
60	6	1-29-66	11,040	-1600	+1.6	124.5	28.8	32.2	1.63	0
60	13	1-29-66	11,055	-1200	-3.0	162.0	25.1	28.9	1.91	0
60	16	1-29-66	11,075	- 850	-2.8	201.0	21.2	28.5	1.90	0
60	17(3)	1-29-66	11,050	- 900	0	202.0	21.7	36.5	2.94	0
62	6	2-1-66	11,075	-1300	+0.7	260.0	11.9	32.0	2.12	0
62	7	2-1-66	11,050	-1200	+1.7	270.0	12.8	29.9	1.90	0
63	5	2-2-66	11,080	- 900	-0.7	29.3	25.4	22.0	2.27	0
63	6	2-266	11,060	- 850	+1.0	31.5	24.5	24.0	2.16	0
65	4	2-6-66	10,155	- 650	0	47.3	18.9	16.8	2.42	0
65	5	2-6-66	10,095	- 650	-1.6	46.5	18.6	15.4	2.29	0
65	11	2-6-66	10,075	- 400	+1.4	107.5	33.6	30.2	2.30	1.20
66	9	2-7-66	10,065	- 650	+3.5	84.6	31.7	31.8	1.66	0.89
66	12	2-7-66	10,105	- 300	0	144.0	31.6	31.4	1.76	0.89
67	8	2-10-66	10,055	-1000	0	213.0	19.6	29.9	1.93	1.10
68	9	2-12-66	10,115	-1200	+2.9	274.0	8.9	30.8	1.49	1.03
70	10	2-16-66	9155	- 700	-3.3	96.0	9.2	15.0	2.27	0
71	5	2-17-66	9065	- 500	-2.0	126.0	10.4	23.1	1.69	0
73	8	2-18-66	11,075	- 100	-0.4	319.5	3.5	30.8	1.55	0

NOTES:

(1) DATA OBTAINED FROM OSCILLOGRAPH ONLY

(2) DATA OBTAINED FROM PHOTOGRAPHIC ANALYSIS ONLY

(3) HARD LANDING - HELICOPTER INSPECTED

(4) ACTUAL ENGINE FAILURE