Report No. NA-67-1
(DS-67-23)

45

FINAL REPORT

Project No. 540-008-01X

AN ANALYSIS OF THE HELICOPTER HEIGHT VELOCITY DIAGRAM INCLUDING A PRACTICAL METHOD FOR ITS DETERMINATION

FEBRUARY 1968

DEPARTMENT OF TRANSPORTATION
Federal Aviation Administration
National Aviation Facilities Experimental Center
Atlantic City, New Jersey 08405

36

DISCLAIMER NOTICE
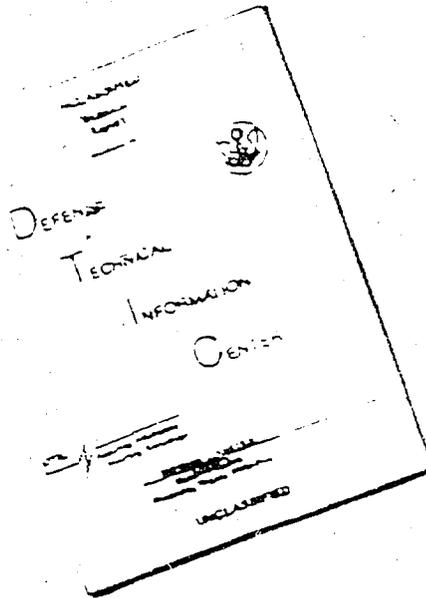
THIS DOCUMENT IS BEST QUALITY AVAILABLE. THE COPY FURNISHED TO DTIC CONTAINED A SIGNIFICANT NUMBER OF PAGES WHICH DO NOT REPRODUCE LEGIBLY.

This Document Contains Missing Page/s That Are Unavailable In The Original Document

THIS DOCUMENT CONTAINED BLANK PAGES THAT HAVE BEEN DELETED

REPRODUCED FROM BEST AVAILABLE COPY

FINAL REPORT

AN ANALYSIS OF THE HELICOPTER HEIGHT VELOCITY DIAGRAM
INCLUDING A PRACTICAL METHOD FOR ITS DETERMINATION

Project No. 540-004-01X
Report No. NA-67-1
(DS-67-23)

Prepared by:

WILLIAM J. HANLEY
GILBERT DEVORE

for

AIRCRAFT DEVELOPMENT SERVICE

FEBRUARY 1968

ABSTRACT

A composite summary analysis was made of the height-velocity (H-V)
diagram test data obtained from the flight testing of three single
engine, single rotor helicopters of varying design characteristics
and basic parameters. The purpose of this analysis was to ascertain
if a practical method for the determination of the H-V diagram could
be evolved, as well as a means to determine the effects of aircraft
weight and altitude on the H-V diagram. Analysis disclosed that H-V
diagrams can be developed for any conventional single rotor helicopter
by the flight test determination of a single maximum performance
critical speed (V_{cr}) point in conjunction with the use of a non-
dimensional curve and the solution of specific key point ratios which
are set forth in the report. An evaluation of the H-V diagram key
point relationships is presented followed by a discussion of the
observed factors affecting autorotative landing following power
failure. A suggested step by step procedure for flight manual type H-V
diagrams is also presented.

iii

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF TABLES

INTRODUCTION

Purpose

The purpose of this project was to determine the effects of aircraft weight and altitude on the basic Height Velocity (H-V) diagrams of three helicopters, and to ascertain if a practical method could be developed for determining the family of curves by analysis rather than by extensive flight testing.

Background

A long-range flight test program was initiated by the Aircraft Development Service, Federal Aviation Administration (FAA), to determine the effects of altitude and weight on the helicopter H-V diagram through actual flight tests. The program was established in response to a need to obtain a practical approach for the determination of the effects of altitude and weight on the helicopter H-V diagram, and for safe autorotational landing criteria in general. In this study only the low speed portion of the H-V diagram was considered. It was felt that the high speed portion involved other considerations that were not pertinent to this analysis and the high speed portion was considered to be not as important as the low speed portion. Within the framework of this program, the testing of three different helicopters possessing widely varying parameters was accomplished in individual but objectively related projects. The results of these tests were published in References 1, 2, and 3.

The H-V envelope for each of the helicopters tested was found to vary as a function of gross weight and density altitude. The similarity of the effects of weight and altitude on the H-V diagrams of the helicopters tested, and the accumulation of helicopter autorotational landing characteristics data warranted an attempt to correlate all the facts. The results of this effort are presented in this report which:

1. Evaluates and summarizes the significant findings of three helicopter H-V flight test projects as individually reported in the FAA Technical Reports, Nos. ADS-1, ADS-46, and ADS-84.

2. Presents empirical factors and a method for the determination of the H-V diagram of conventional single rotor helicopters.

3. Discusses observations of the factors which influence the determination of the critical velocity of the H-V diagram and autorotational landing characteristics in general.

1

DISCUSSION

Test Aircraft

The three helicopters tested under the long-range program were all single engine, single rotor helicopters as shown in Fig. 1. Helicopter No. 1 was a lightweight model with relatively low disk loading and moderate rotor inertia. On the other hand, Helicopter No. 2 was also a lightweight model but with a comparatively high disk loading and a low rotor inertia. Helicopter No. 3 was a large, heavyweight helicopter with high disk loading and a comparatively high rotor inertia. Tabulated in Table I are some of the basic parameters which further define the three helicopters and which were involved in the analytical study that was the basis of this report. It was considered that a representative cross section of helicopters was chosen with respect to design considerations as they might influence the test results. Of the three helicopters tested, only Helicopter No. 1 had an altitude rated powerplant system while Helicopters Nos. 2 and 3 essentially had sea level rated systems; consequently, tests at high airport elevations with the latter two helicopters could not be conducted at all the desired test weights because of the power-on performance characteristics. More detailed specifications on the test helicopters are presented in References 1, 2, and 3 which are the basic references of this report.

TABLE I

TEST AIRCRAFT CHARACTERISTICS

	Unit	Helicopter No. 1	Helicopter No. 2	Helicopter No. 3
Max. Cert. Gross Weight	lbs	2850	1670	13,000
hp/rpm @ S.L. continuous	hp/rpm	220/3200	160/2700	1275/2500
Disk Loading W/A	lb/ft^2	2.63	3.32	5.28
Rotor Solidity Ratio, σ		.0314	.0424	.059
Rotor Inertia, I_R	Slug ft^2	710	140	5800
No. of Rotor Blades		2	3	4
Landing Gear Config.		Skid	Skid	Wheel
Rotor Config.		Teetering	Articulated	Articulated
* Equiv. Flat Plate Area	ft^2	18	14.5	36.5
Rotor Disk Area, A	ft^2	1083	503	2460
Rotor Tip Speed, $R\Omega$	ft/sec	690	640	700

*Frontal

HELICOPTER NO. 1
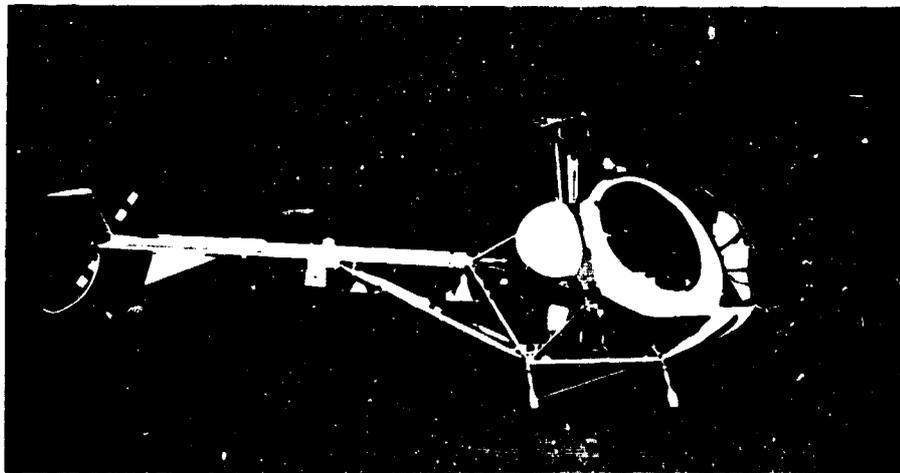
HELICOPTER NO. 2

HELICOPTER NO. 3

FIG. 1   TEST AIRCRAFT

3

Test Instrumentation

Airborne and ground instrumentation was utilized to record helicopter performance, aircraft space position and meteorological information. In general, the same parameters were measured with identical or similar instrumentation for all three flight test projects.

Test Procedures

1. Basic Test Methodology

Early in the planning stages, certain basic ground rules were established as mandatory and applicable throughout all the project tests. Essentially, these guidelines were as follows:

a. Maximum Performance

The H-V diagrams developed had to be representative of a maximum performance effort; that is, it was necessary to extract the maximum capabilities from the helicopter during each and every test point throughout the entire program. This was necessary since there were no known parameters available by which a fixed degree of conservatism in obtaining these test points could be determined or measured. Extraction of the maximum performance capability of the aircraft is dependent upon the skill of the pilot. For this reason, professional engineering test pilots, thoroughly familiar with the helicopter each flew, and well skilled in the mechanics of determing H-V diagrams, were employed for the piloting function.

b. Test Weights and Elevations

The weights and elevations selected for each of the helicopters tested were chosen with sufficient spread to clearly indicate the effects of these variables on the H-V diagram. The test sites selected provided a broad range of density altitudes and the incremental weight changes averaged approximately 8% of maximum certificated gross weight. Table II lists the test site elevations and helicopter weights used in the program.

TABLE II							
TEST WEIGHTS AND ELEVATIONS							
Elevation (ft) MSL	Aircraft Test Weight (lbs)						
	Helicopter No. 1 2615 2850 2850			Helicopter No. 2 1450 1600		Helicopter No. 3 9100 10,100 11,100		
-117						X	X	X
332	X	X	X	X	X			
4118	X	X	X	X	X	X	X	X
6263						X	X	
7120	X	X	X	X				
9870	X	X						

X Indicates site elevation/weight combination for flight tests.

Note: Density Altitudes at time of tests were generally higher than
field elevations.

c. Flight Entry Conditions

All of the simulated power failure landing maneuvers had to
be executed from a trim, steady state, level flight entry condition in
order to obtain repeatability and accuracy. The use of an accelerated
climbout technique along the lower boundary was deemed to include too
many variables to provide the accuracy required for the basic
determination. Admittedly, the accelerated climbout technique would
have provided a more realistic approach, but in addition to the
requirements for repeatability, safety as well as the economics of
the program also had to be considered. It was, therefore, decided that
the level flight technique was the best compromise.

For test points along the upper boundary, collective pitch
reduction was applied with a one-second delay after simulated power
failure as well as with no-delay. For all test points initiated along
the lower boundary, collective pitch reduction, when applied, was
immediate.

d. Touchdown Speeds

There were no limitations placed upon the pilots with respect
to touchdown speeds. They were allowed to contact the runway at whatever
forward velocity was required in order to fulfill the requirements of
extracting the maximum capability of the aircraft for every landing.

SUMMARY OF RESULTS

Scope

An analysis of the test data of the total program was undertaken to determine if sufficient data existed to permit the formulation of an empirical and/or theoretical method of predicting the size and shape of the H-V diagram for conventional single rotor helicopters.

Height-Velocity Diagrams

1. General

The height-velocity diagrams developed for each helicopter exhibited remarkable similarity in shape to each other. Each helicopter tested produced a family of H-V diagrams which reflected the effects of weight and altitude, and each of these families was similar to the families of the other helicopters not only in shape but in the relationship of all the key points.

This similarity existed for the conditions of both no-delay and one-second delay in pitch reduction following simulated power failure. A typical example of H-V diagrams developed in this program is shown in Figure 2, on which the basic key points are designated. The fact that a family of curves existed that were so similar gave rise to the concept of a nondimensional curve that could be utilized to facilitate computation of H-V diagrams.

2. Generalized Height-Velocity Curve

Each of the helicopters tested produced a family of diagrams which were related as functions of weight and altitude; therefore, the family of diagrams for each helicopter could be presented in the form of one nondimensional H-V curve. This procedure was first suggested in Reference 4, and was more recently accomplished in Reference 5, utilizing the data developed in one of the tests of this program. The H-V families of all three helicopters have been further combined in this summary analysis to form one composite nondimensional H-V curve. The composite nondimensional H-V curve was generated by computing the key point ratios V_n/V_{cr}, h_1, and h_2 as shown on Figure 3. All of the H-V diagrams developed in this program were nondimensionalized in accordance with the procedure shown on Figure 3 and the resulting nondimensional points were plotted on Figure 4.

A mean nondimensional H-V curve was then constructed through the composite data. A table has been prepared which presents the coordinates of this mean H-V curve. It is thus possible to construct an H-V diagram for any weight (minimum operating to maximum gross) and altitude (8,000 ft maximum) when one of the key points, h_{min} or V_{cr}, is known by reversing the above procedure. Utilizing the mean values of the chart on Figure 4, the ratios can be solved for values of h_n and V_n.

6

FIG. 2   TYPICAL H-V DIAGRAMS

7

NON-DIMENSIONALIZING PROCEDURE

1. NON-DIMENSIONAL VELOCITY

AT A SELECTED VELOCITY VALUE V_n, DIVIDE BY VALUE V_{cr} TO OBTAIN NON-DIMENSIONAL VELOCITY

2. NON-DIMENSIONAL HEIGHT

a.–LOWER BOUNDARY OF H-V DIAGRAM FOR VELOCITY V_n DETERMINE NON-DIMENSIONAL HEIGHT h_1 BY RATIO SHOWN IN FIGURE; WHERE h_{n_1} EQUALS HEIGHT OVER GROUND AT VELOCITY V_n; AND h_{n_1} AND h_{cr} ARE KNOWN VALUES AS DEFINED IN THE GLOSSARY

b.–UPPER BOUNDARY OF H-V DIAGRAM FOR SAME VELOCITY V_n DETERMINE NON-DIMENSIONAL HEIGHT h_2 BY RATIO SHOWN IN FIGURE; WHERE h_{n_2} EQUALS HEIGHT OVER GROUND AT VELOCITY V_n AND h_{min} AND h_{cr} ARE KNOWN VALUES AS DEFINED IN THE GLOSSARY

3. NON-DIMENSIONAL CURVE

PLOT NON-DIMENSIONAL HEIGHTS AND VELOCITIES THUS OBTAINED AS SHOWN ON FIG. 4

TYPICAL H-V DIAGRAM

$$h_2 = \frac{h_{min} - h_{n_2}}{h_{min} - h_{cr}}$$

$$h_1 = \frac{h_{n_1} - h_{max}}{h_{cr} - h_{max}}$$

VELOCITY MPH

HEIGHT OVER GROUND, FEET

FIG. 3   METHOD OF CONVERTING TYPICAL H-V DIAGRAM
INTO CALCULATED KEY POINT RATIOS

△ HELICOPTER NO. 1
□ HELICOPTER NO. 2
○ HELICOPTER NO. 3

MEAN CURVE VALUES		
V/V_{CR}	h_1	h_2
0	0	0
.10	0	.07
.20	.005	.13
.25	.01	.17
.30	.02	.20
.35	.025	.24
.40	.035	.28
.45	.05	.31
.50	.06	.35
.55	.08	.39
.60	.10	.43
.65	.13	.48
.70	.16	.53
.75	.20	.58
.80	.25	.63
.82	.27	.66
.84	.30	.68
.86	.33	.70
.88	.37	.73
.90	.40	.76
.92	.46	.79
.94	.52	.82
.96	.60	.85
.98	.68	.90
.99	.80	.93
1.00	1.00	1.00

FIG. 4   COMPOSITE NONDIMENSIONAL CURVE

Essentially, h_{min} and V_{cr} are the only unknowns. As will be shown later in this report, h_{cr} and h_{max} have been established by this program and h_{min} is empirically related to V^2_{cr}. It is necessary to test for only one of the two values (i.e., h_{min} or V_{cr}) to develop the H-V diagram. Since tests for h_{min} have been shown to be less accurate and more difficult to conduct than tests for V_{cr}, it would be more logical to test for V_{cr}.

3. Evaluation of Key Point Relationships

a. The Critical Speed Versus Weight

Each helicopter tested showed a linear relationship of V_{cr} versus weight over the range of weights and altitudes at which each were tested. Since only the helicopter utilized in Reference 1 was equipped with an altitude engine, it was the only helicopter which could be tested over a full weight range at the highest altitudes tested. There was no evidence of drag divergence or detrimental blade stall which would cause a serious alteration of the linear relation for an altitude range of at least sea level to 8000 feet. This was true for the one-second delay as well as the no-delay test results. In each of the three reports, an expression as shown below was presented from which V_{cr} for a new weight could be determined from a V_{cr} obtained at some test weight.

$$V_{cr} = V_{cr_{test}} + C_1 \triangle W \quad \text{where } C_1 = \frac{dV_{cr}}{dW}$$

Analysis of the three sets of test data revealed that if the experimentally determined values of C_1 were multiplied by rotor area, the products were essentially constant; i.e.,

$$C_1 A = 22.6$$

The use of this empirically established relationship in the previous equation involving V_{cr}, results in an expression for the change in V_{cr} due to a weight change in terms of a delta disk loading; i.e.,

$$V_{cr} = V_{cr_{test}} + 22.6 \frac{\triangle W}{A}$$

so that a new V_{cr} can be readily obtained by applying an average value of $C_1 A$ (22.6) to the delta disk loading.

Figure 5 was developed using the average test value of $C_1 A = 22.6$. From this chart C_1 can be determined directly as a function of rotor disk area. Because of the representative, but widely varying basic parameters of the three helicopters tested, it is concluded that all conventional single-rotor helicopters will fit the experimentally determined value of 22.6 for $C_1 A$.

10

FIG. 5 C₁ VERSUS DISK AREA

LEGEND

SYMBOL	HELICOPTER	C_1	A	C_1A
◉	NO. 1	.02075	1083	22.48
▲	NO. 2	.04500	503	22.64
▥	NO. 3	.00925	2460	22.75

DISK AREA ~ 100 SQUARE FEET

C_1 ~ MILES PER HOUR PER POUND AIRCRAFT WEIGHT

11
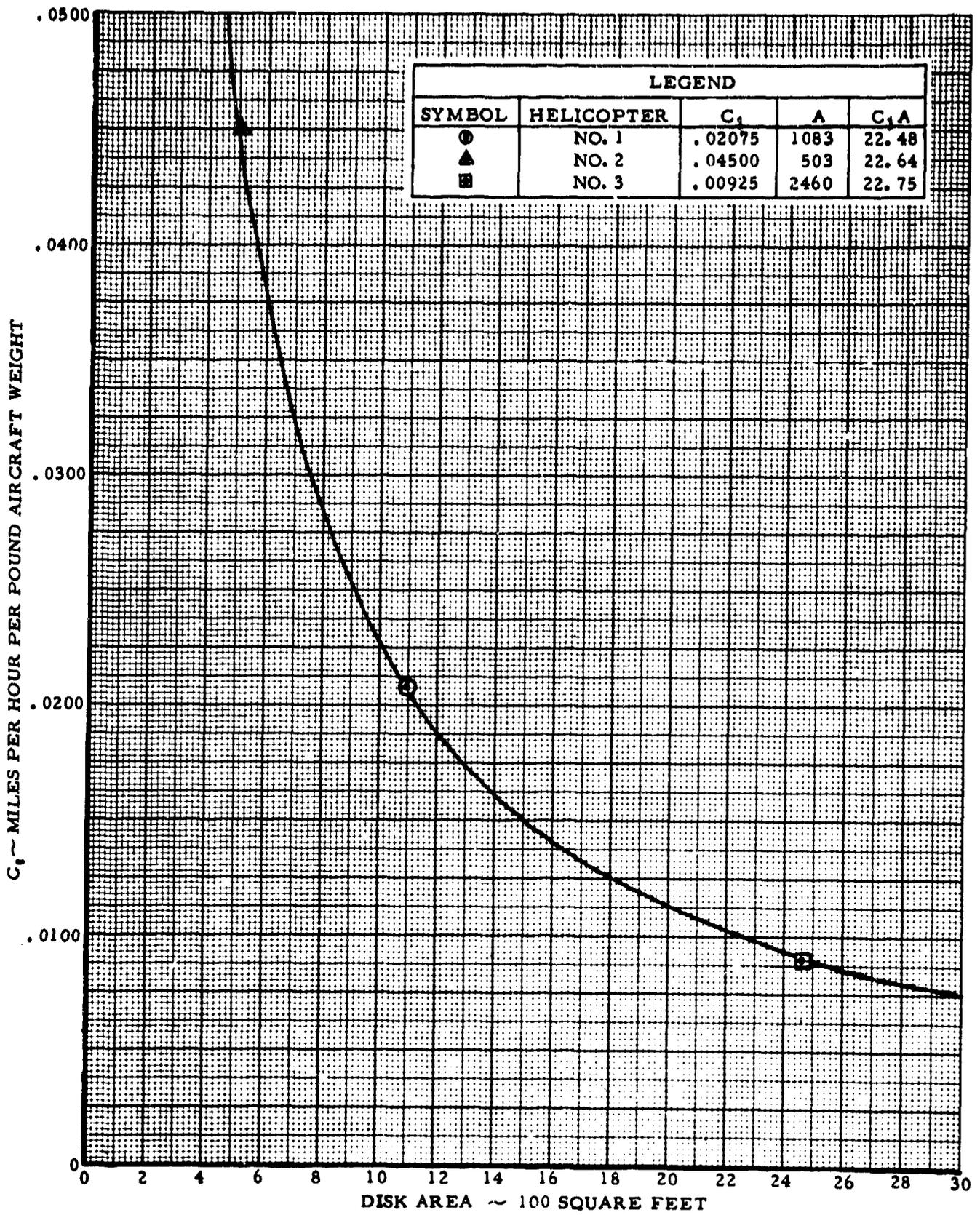
b. The Critical Speed Versus Altitude

The basic purpose of this portion of the composite analysis was to establish the effects of altitude on the critical speed. The three project tests had proven rather conclusively that the variation of V_{cr} with altitude was linear and was valid over a normal altitude range of at least sea level to 8000 feet. The helicopters which were limited in power available could only be tested at lighter weights at the higher altitudes, but there was no evidence to suspect that for the normal altitude range there would be any deterioration from the linear characteristics. As in the case of V_{cr} versus weight, the linear relationship of V_{cr} versus altitude was maintained for both the no-delay and one-second delay test results. An increase in V_{cr} was experienced for a one-second delay and the increase was constant with altitude changes such that the linearity of V_{cr} versus altitude was not affected. No empirical solution was found for determining the change in V_{cr} due to a change in altitude on a unified basis involving all three helicopters as was done in the case of V_{cr} versus weight. In each of the three reports an expression as shown below was presented from which V_{cr} for a new altitude could be determined from a V_{cr} obtained at some other test altitude.

$$V_{cr} = V_{cr_{test}} + C_2 \, \Delta H_D \quad \text{where } C_2 = \frac{dV_{cr}}{d H_D}$$

The values of C_2 obtained for each helicopter could not be correlated or combined with known parameters to arrive at a solution which was applicable to the three helicopters for the determination of V_{cr} at a different altitude, as was done in the case of V_{cr} versus weight. On a practical basis, however, the variation of C_2 values determined for the three test aircraft are not so great as to preclude a working solution from the data obtained. The C_2 values of the helicopters tested varies between 1.6 and 2.5 mph per thousand feet, with Helicopters Nos. 2 and 3 being 2.5 and 2.3 mph per thousand feet respectively. It should be pointed out that the helicopters tested encompassed an extreme range of the major parameters and therefore can be considered to encompass helicopters whose parameters fall within this range. It is not likely, therefore, that the value of C_2 (dV_{cr}/dH) for conventional single rotor helicopters will exceed the maximum value obtained in this program. Until such time as additional information is obtained to establish an empirical or theoretical relationship for C_2, the use of the maximum value of 2.5 mph per thousand and feet of altitude is a practical solution.

c. The Critical Height

The critical height, h_{cr}, was found to be reasonably constant over the altitude and weight range for all three helicopters and averaged approximately 100 feet. The critical height for Helicopter No. 1 (Reference 1) varied between 90 and 100 feet and was assumed to be constant at 95 feet. As a result of the tests of Reference 1, Helicopter

No. 2 was tested more critically in this area to determine if a variation existed in h_{cr}. This helicopter showed a variation in h_{cr} from 80 to 100 feet over the range of weights and altitudes tested. The diagrams of Helicopter No. 2 had originally been prepared with a pronounced "chin" or distention of the lower boundary. As a result of this analysis; i.e., analyzing the curves of all three helicopters collectively, it was determined that they could have been faired otherwise, which would result in not only reducing the chin but also in raising the critical height as well. Helicopter No. 3 showed a range of critical heights between 90 and 110 feet. Further analysis of all of the data indicated that at maximum gross weight at sea level, h_{cr} can be defined as approximately 100 feet. For lighter weights at sea level, h_{cr} would be somewhat lower and for higher altitudes at maximum gross weight, h_{cr} would be somewhat higher. Thus, with a weight reduction at altitude, h_{cr} would remain approximately constant at 100 feet. This finding was important to the determination of an H-V diagram because it permits the height at which V_{cr} occurs to be established.

d. Critical Speed Squared Versus High Hover Height

In each of the three previous reports a linear relation was found to exist between the high hover height, h_{min}, and the square of the critical speed, V^2_{cr}, as shown in Fig. 6. Since this appeared to be significant with respect to the H-V diagram, an analysis of this particular relationship was made for the three helicopters.

As there was a slight discrepancy between the h_{min} versus V^2_{cr} curves of the three helicopters, as shown in Fig. 6, the basic test data of all three were examined collectively. This examination led to the conclusion that one common curve could be drawn for all three helicopters. This was of particular significance since it appeared that the relationship of h_{min} and V^2_{cr} was the same for all helicopters independent of the helicopter parameters, and determination of one of these key points could be made if the other were known.

From this straight line common curve shown in Fig. 6, an empirical equation was established as follows: $h_{min} = 200 + .1336 \ V^2_{cr}$.

Conventional single rotor helicopters subscribe to this expression. This relationship, which associates the high hover point with the critical speed, made it possible to define the limits of the upper boundary of the H-V diagram by simply obtaining one of these points.

e. Low Hover Height Versus Weight and Altitude

The landing following power failure from the low hover height, h_{max}, is the one regime of flight on the H-V diagram which lends itself to an energy analysis and this has been effectively treated in Reference 5. A study was made of the low hover height and the lower boundary to obtain a simplified empirical solution. It was determined

13

$$h_{min} = 2 h_{cr} + C_5 V^2_{cr}$$

WHERE h_{cr} = 100 ft AND C_5 = .1336

$$h_{min} = 200 + .1336 V^2_{cr}$$

HELICOPTER NO. 3 (REF.)

HELICOPTER NO. 1 (REF.)

HELICOPTER NO. 2 (REF.)

(V_{cr}) MPH)^2

HIGH HOVER HEIGHT (h_{min}) FEET ABOVE GROUND

500 450 400 350 300 250 200 150

0 200 400 600 800 1000 1200 1400 1600 1800 2000 2200
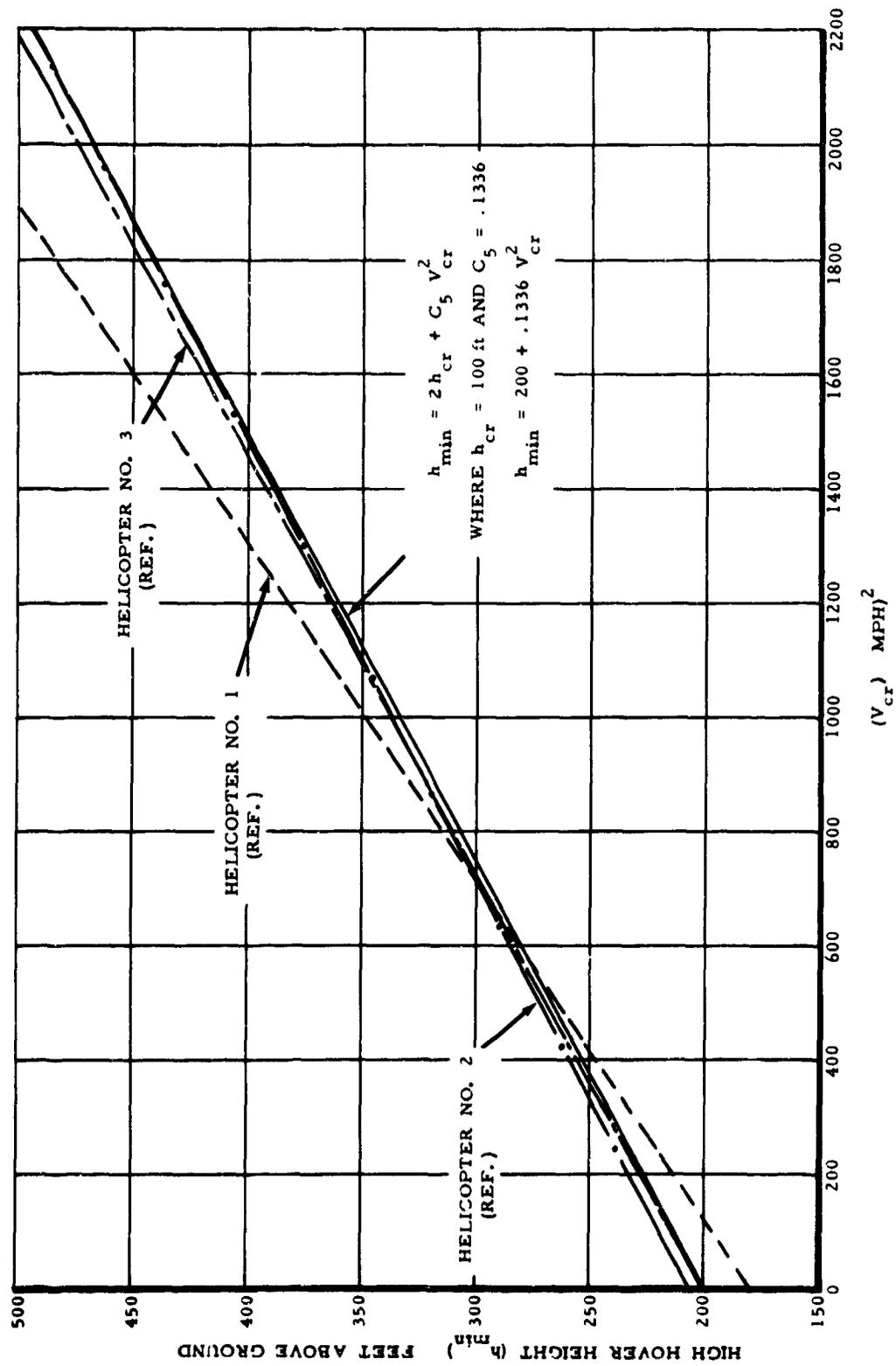
FIG. 6 HIGH HOVER HEIGHT (h_{min}) VERSUS
SQUARE OF CRITICAL VELOCITY (V^2_{cr})

14

that the variation of h_{max} for any of the helicopters was small. Of the three helicopters tested, the largest variation encountered over the weight and altitude range was a variation in h_{max} from approximately 5 to 20 feet. The 10-foot height, generally accepted in lieu of conducting h_{max} tests, appears to be a realistic value for h_{max} at sea level and maximum gross weight. In the interest of conservatism, simplicity, and for purposes of maintaining the family relationship of H-V diagrams, it is recommended that a reduction of 1 foot of height in h_{max} per 1000 feet increase in density altitude be appropriately applied to the basic 10-foot sea level value. Similarly, for a weight reduction from maximum gross weight to minimum operating weight, h_{max} should be increased 5 feet above the basic 10-foot sea level value. The value of h_{max} thus derived provides the lower limit of the lower boundary portion of the H-V curve.

Factors Affecting the Autorotative Landing Following Power Failure

1. Rotor Inertia and Rotor Speed

Two of the helicopters tested had a normal operating rotor speed which produced tip speeds of approximately 700 ft/sec. The lightweight helicopter rotor speed was on the low side with a tipspeed of 640 ft/sec. All three helicopters employed metal blades of the NACA 00 series and thus operated on approximately the same profile drag coefficient versus mean lift coefficient curve (δ versus C_L). The profile drag, therefore, was essentially the same since all three helicopters operated at approximately the same mean lift coefficient.

An investigation was made into the relative rotor speed decay to determine the effect of rotor inertia on rotor speed decay following throttle chop along the upper boundary. In order to compare the decay trends of the three helicopters, a plot of rotor speed in percent of rated rpm versus time was constructed as shown on Figure 7. It is interesting to note that the helicopter with the largest inertia per pound of aircraft weight (Helicopter No. 3) exhibits the largest percent decay with time. The data show that the decay rate relationship between the helicopters remains essentially the same whether a one-second delay or no-delay was employed following throttle chop prior to collective pitch reduction, although the percent decay is largest after a one-second delay. It should be noted also that following a one-second delay, the helicopter with the least inertia per pound of aircraft weight does not have the highest V_{cr} (sea level, maximum gross weight) and conversely, the helicopter with the most inertia per pound does not have the lowest V_{cr}. Thus it was not possible to establish any pattern of the effect of rotor inertia on V_{cr}, and other factors are obviously present in the development of V_{cr}.

Only a limited amount of testing was conducted in Reference 1 to determine the effects of added rotor inertia on the location of V_{cr} or the upper boundary of the H-V diagram. From this limited data the effects of inertia appeared minimal. Likewise, the test helicopters

15
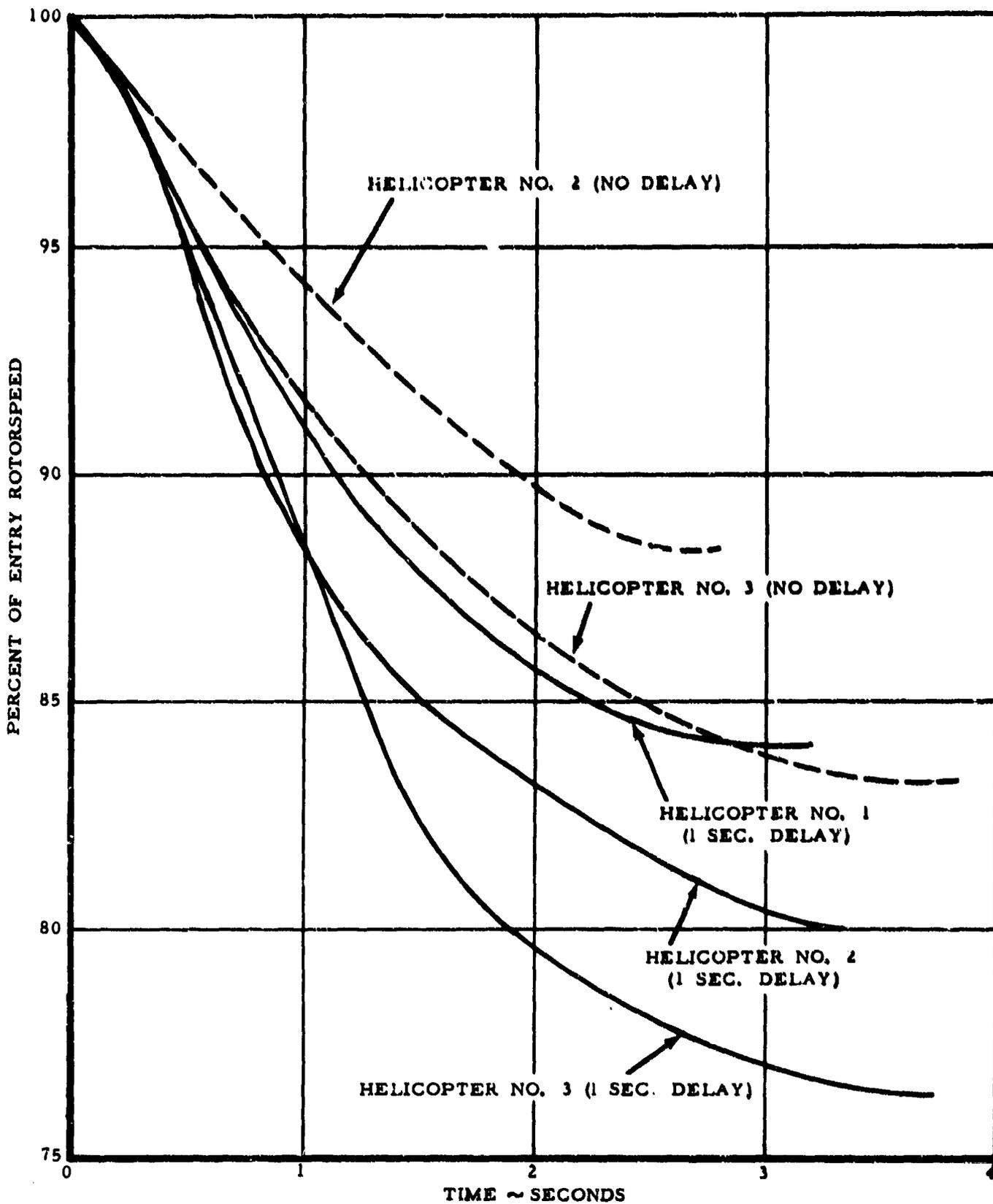
FIG. 7   AVERAGE ROTOR SPEED DECAY TRENDS
FROM HIGH HOVER ENTRY POINTS

16

having varying inertia values per pound of aircraft weight from .1 to
.6 displayed a consistent pattern in the h_{min} - V^2_{cr} relation and along
the upper boundary. From high hover to V_{cr}, therefore, it is evident
that there are factors other than rotor inertia which determine the
values of h_{min} and V_{cr}.

2. Blade Stall Considerations

During the tests conducted on Helicopter No. 1, no comment
was made by the pilot with respect to blade stall. There appeared to
be no evidence of blade stall at any weight or altitude. The pilot
never indicated that blade stall was a consideration. Blade stall did
not become a consideration until the last test project. At the higher
weights and generally throughout the altitude range during the tests of
Reference 3, the pilot found it necessary to adjust his technique to
compensate for the effects of blade stall during flare and final
collective pull-up. There were occasions during the second helicopter
test program when the pilot reported "falling through" the flare.
Although he did not report it as blade stall, it is conceivable that
blade stall was a factor in "falling through" the flare. It appears
that blade stall as a factor in the autorotative landing following power
failure is a function of disk loading. Helicopter No. 1, which had a
relatively light disk loading, did not require high collective blade
angles to effect its landings and this, undoubtedly, contributed to the
lack of blade stall. On the other hand, Helicopter No. 3 with its
extremely high disk loading needed all the blade angle possible which in
turn contributed to blade stall as a factor. It is also possible that
rotor inertia is an influencing factor in that the higher inertia rotor
regains its rpm less rapidly in the flare, whereas, low inertia rotor
regains its rpm more rapidly. Reference to Figure 7 shows that loss of
rotor speed following throttle chop is a function of disk loading and
not rotor inertia.

3. The Maneuver - Cyclic and Collective Flare

Analysis of the time histories of all three helicopters showed
that the cyclic and collective controls were utilized in similar fashion
and produced traces of pitch attitude, acceleration, swashplate angle,
and blade angle which were repetitious in pattern. This can be seen in
Figures 8 through 10, which show sample time histories of the three
aircraft for the three basic H-V diagram areas - high hover, low hover,
and V_{cr}. The deceleration following power failure along the upper
boundary from h_{min} to V_{cr}, for example, consistently showed approximately
.75g from 1g steady level flight. There was clear evidence in the time
histories that when power failure occurred along the upper boundary, the
cyclic flare was the most important control factor in executing a power-
off landing. While the collective pitch was vital to reducing the load

17
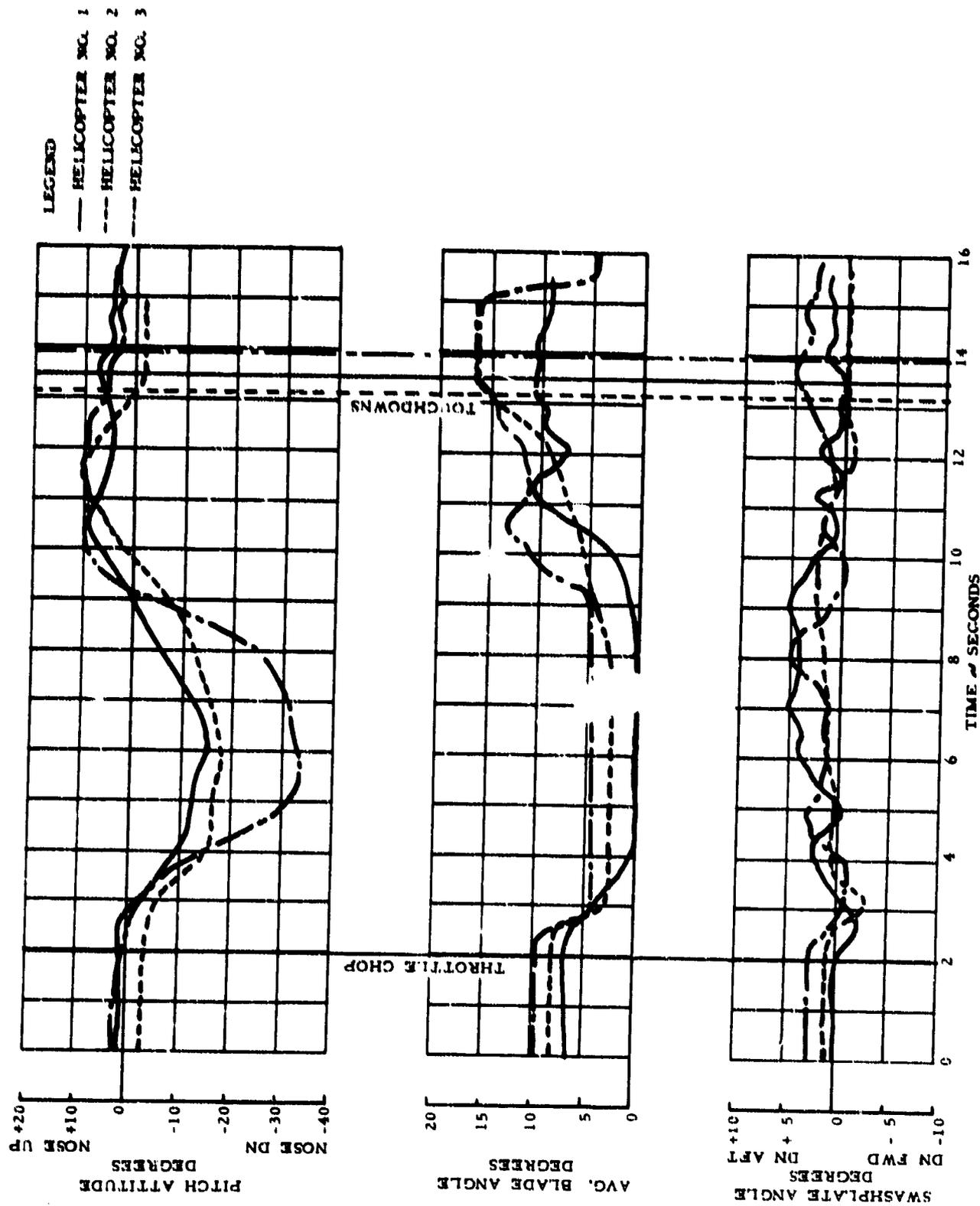
FIG. 8 COMPARISON OF TIME HISTORY DATA FOR HIGH
HOVER POINTS - THREE TEST HELICOPTERS

18
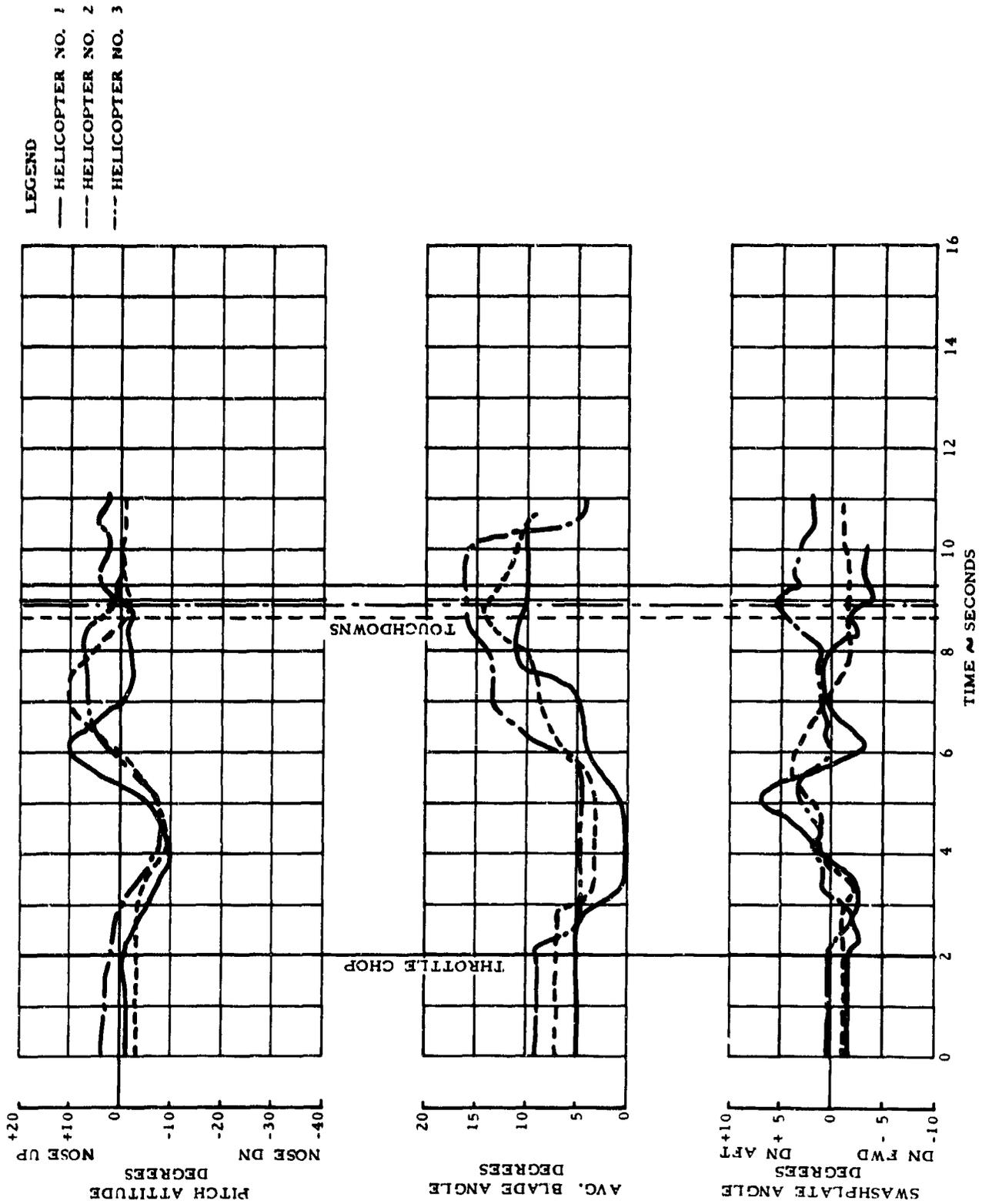
FIG. 9 COMPARISON OF TIME HISTORY DATA FOR CRITICAL SPEED AREA (V_{cr}, h_{cr}) - THREE TEST HELICOPTERS

19
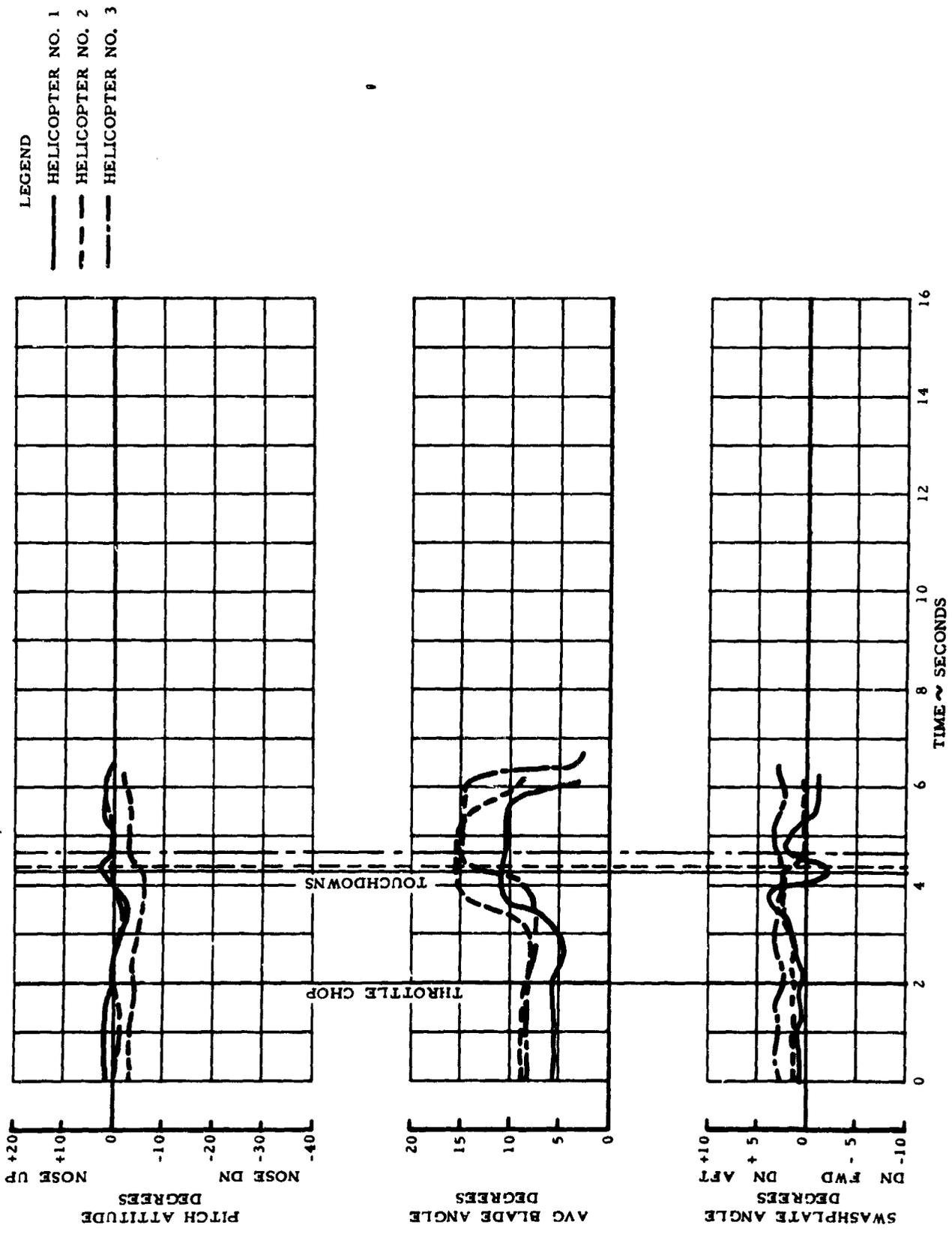
FIG. 10 COMPARISON OF TIME HISTORY DATA FOR LOW HOVER POINTS - THREE TEST HELICOPTERS

20

factor on contact, it was the cyclic flare that generated the rotor speed and rate of descent reduction which made a good collective flare possible. Without this technique, significantly greater airspeed was required prior to contact for a good collective flare.

The landing techniques employed for the three helicopters were reasonably consistent, even though Helicopters Nos. 1 and 2 had skid gears and Helicopter No. 3 had a tail wheel/mainwheel configuration. In the case of Helicopter No. 3, the technique was basically the same as that employed for the skid gear helicopters even though the technique did not call for leveling the helicopter before contact. In landing Helicopter No. 3 it was necessary to apply only partial collective pitch to cushion the tailwheel while "feeling for the ground." The remaining collective application was applied to cushion the main gear touchdown which determined the acceptable load factors. The helicopter was landed, therefore, with a technique midway between an air rotation for level landing (as were the skid gear aircraft) and a pure flare landing (as would be done for a main wheel aft configuration).

Referring again to Figures 8 through 10, it can be seen that all the essential elements of control and aircraft motion are similar for the three aircraft. During the tests and subsequent data analysis of Helicopter No. 2, it was believed that distention of the lower boundary occurred because of the low rotor inertia. There was insufficient time from throttle chop to touchdown to regain rotorspeed through a cyclic flare, and therefore, the landings had to be made essentially by means of a collective flare. Subsequent testing on Helicopter No. 3, however, showed that the same condition existed on the high inertia rotor. Points developed along the lower boundary with Helicopter No. 3 were also accomplished with minimum cyclic flare and maximum collective flare. The time from throttle chop to touchdown was insufficient for the high inertia rotor to build back any rotor speed, so that essentially, the techniques employed were consistent for all helicopters. It is the time element between throttle chop and touchdown, established by the entry altitude and speed, which dictates the type of maneuver necessary along the lower boundary. Rotor inertia appeared to have a very small influence on this regime of flight.

Method for Converting Test Program H-V Diagrams to Flight Manual Type H-V Diagrams

The results of the tests conducted in this program established that along the upper boundary the use of a one-second delay simply added an increment of speed such that the upper boundary was displaced laterally. The one-second delay diagrams complied with the patterns established in the Height Velocity Diagram section of this report and were, therefore, subject to the nondimensionalized form. Since the upper boundary of the flight manual type H-V Diagram is determined utilizing

21

a one-second delay, it is apparent that the curves should be identical except for a margin of conservatism to differentiate between the maximum performance data obtained by professional pilots in this program and the capabilities of the average pilot. If an appropriate margin is thus applied to V_{cr}, a flight manual type H-V diagram upper boundary can be developed by a single point determination of a maximum performance V_{cr} at an h_{cr} of 100 feet. Furthermore, since we have nondimensionalized the H-V diagram, the lower boundary when developed from the same V_{cr} will provide a lower boundary that will be equivalent to the accelerated climbout technique generally utilized in establishing the lower boundary. In this regime a one-second delay is not employed; therefore, the two increments in V_{cr} from no-delay to one-second delay, and from one-second to the percent margin increase, provides the necessary margin for the accelerated climbout lower boundary.

Procedures for Obtaining Flight Manual Type H-V Diagrams for Range of Weights and Altitudes

A procedure for developing a flight manual type H-V diagram established in accordance with the data and determinations contained in this report would make it possible to obtain a set of H-V diagrams over a range of weights and altitudes from a single test to determine V_{cr}. The following procedure, therefore, is considered reasonable for establishing such a flight manual type H-V diagram.

1. At a given gross weight (near maximum), and at a given density altitude(near sea level), and with a neutral c.g., determine a maximum performance V_{cr} with a one-second delay at an h_{cr} of 100 feet.

2. Using Figure 5, determine C_1 and use the equation $V_{cr} = V_{cr_{test}} + C_1 \Delta W$ to establish V_{cr} for maximum gross weight. Likewise, using the value of C_2 of 2.5 mph per 1000 feet, correct V_{cr} for sea level using the equation $V_{cr} = V_{cr_{test}} + C_2 \Delta H_D$. A V_{cr} for maximum gross weight at sea level has now been established.

3. Add to the V_{cr} established in Item 2, above, an increment of speed(in mph) to provide an adequate safety margin for the average pilot.

4. Using the equation $h_{min} = 200 + .1336 \, V^2_{cr}$, determine h_{min}. Select an h_{max} of 10 feet an h_{cr} of 100 feet.

5. With V_{cr}, h_{cr}, h_{min}, and h_{max} now known, establish the H-V diagram using the nondimensional data from the mean curve chart of Figure 4 and the equations for h_1, h_2, and V_n/V_{cr} shown on Figure 3. A flight manual type H-V diagram for maximum gross weight at sea level has now been established.

22

6. Using C_1 and C_2 establish V_{cr} for all desired weights and altitudes.

7. Determine h_{min} as in Step 4. for each V_{cr}.

8. Determine h_{max} by subtracting 1 foot per 1000 feet of density altitude from the basic 10 feet for sea level maximum weight and adding 5 feet over the weight range from maximum weight to minimum operating weight.

9. Determine h_{cr} by adding 1 foot per 1000 feet of altitude to the basic 100 feet at constant weight and decrease 10 feet equally spaced over the operating weight range for constant altitude.

10. Knowing all key points, repeat as in Step 5. and obtain a family of H-V diagrams over the desired weight and altitude range.

11. Spot check the diagrams with average pilot technique at critical c.g.

12. Establish a procedure for utilizing the diagrams thus developed that is appropriate to the particular helicopter in accordance with its operating weight range and its hovering performance capabilities.

CONCLUSIONS

Based upon an analysis of the data obtained in this program and an analytical study of the results in terms of basic helicopter parameters, it can be concluded that

1. A practical method has been evolved for determining a family of flight manual type H-V diagrams for any single engine, single rotor helicopter over a range of weights from maximum to minimum operating, and altitudes from sea level to 8000 feet, by utilizing a single maximum-performance test point determination of V_{cr}.

2. The H-V diagrams of any single rotor helicopter form a family of curves which are defined by a set of linear equations involving the key points of the H-V diagram, V_{cr}, h_{cr}, h_{min}, h_{max}, which vary as follows:

    a. The variation of V_{cr} with a change in weight is a function of disk loading and gross weight.

    b. The variation of V_{cr} with a change in altitude can be accepted for practical purposes as varying 2.5 mph per 1000 feet change of density altitude.

    c. The critical height (h_{cr}) is approximately 100 feet at sea level and maximum gross weight increasing 1 foot per 1000 feet of density altitude and decreasing 10 feet over the weight range from maximum weight to minimum operating weight.

    d. The high hover height (h_{min}) varies linearly as the square of the critical speed and can be expressed in equation form as

    $$h_{min} = 200 + .1336 \, V^2_{cr}$$

    where V_{cr} is in mph

    h_{min} is in ft

    e. The low hover height (h_{max}) is approximately 10 feet at sea level and maximum gross weight decreasing 1 foot per 1000 feet of density altitude and increasing 5 feet over the operating weight range from maximum weight to minimum operating weight.

3. Height-velocity diagrams can be generalized in nondimensional terms.

24

RECOMMENDATIONS

Based upon the results of this summary analysis, it is recommended

1. In the interest of standardization, reduction of flight test risks through better predictions, and a reduction of testing at several altitudes, that consideration be given to using V_{cr} procedures and criteria of this report as guidance material in developing flight manual type H-V diagrams required in the fulfillment of the airworthiness standards for normal category rotorcraft (FAR 27.79).

2. Further study and flight testing be initiated to determine the relationship of V_{cr} with the basic helicopter parameters which may provide a means of H-V diagram determination without the requirement for any flight testing.

REFERENCES

1. Hanley, W. J., and Devore, G., _An Evaluation of the Effects of Altitude on the Height Velocity Diagram of a Single Engine Helicopter_, Technical Report ADS-1, February 1964.

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.

3. Hanley, W. J., Devore, G., and Martin, S., _An Evaluation of the Height Velocity Diagram of a Heavyweight, High Rotor Inertia, Single Engine Helicopter_, Technical Report ADS-84, November 1966.

4. Rich, M. J., _An Energy Absorption Safety Alighting Gear for Helicopter and VTOL Aircraft_, IAS Paper No. 62-16, January 1962.

5. Pegg, R., _Calculating Changes in the Helicopter Height Velocity Diagram with Changes in Density Altitude and Gross Weight_, University of Virginia Master's Thesis, May 1965.

APPENDIX 1

GLOSSARY OF TERMS

GLOSSARY OF TERMS

H-V Diagram: A graphical presentation which defines an envelope of flight with respect to airspeed and height above the ground, within which, in the event of power failure, a safe power-off landing could not be accomplished.

Key Points: The critical points on the H-V diagram which define the size of the diagram. They are h_{min}, h_{max}, and V_{cr}, h_{cr} as defined below.

Upper Boundary: The curve describing that portion of the H-V diagram between the high hover point (h_{min}) and the point for critical speed (V_{cr}, h_{cr}).

Lower Boundary: The curve describing that portion of the H-V diagram between the low hover point (h_{max}) and the point for critical speed (V_{cr}, h_{cr}).

"Knee"   A colloquial term that is synonymous with that curved portion of the H-V diagram in close proximity of the point for critical speed (V_{cr}, h_{cr}).

Conventional
Single Rotor
Helicopter: A helicopter which employs a gear and/or belt system for transmitting power to the main rotor, and does not utilize main rotor tip drive of any kind.

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, ft.

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, ft.

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, ft.

H_D: Density altitude at the point of landing, ft.

h: Height of the helicopter above the ground, ft.

W: Helicopter weight, lb.

GLOSSARY OF TERMS CONTINUED

$\underline{CAS}$: Calibrated airspeed - indicated airspeed corrected for instrument and position error, mph.

$\underline{C_1}$: The slope of the linear relationship of the change in critical speed with change in weight $= \dfrac{dV_{cr}}{dW_D}$

$\underline{C_3}$: The slope of the linear relationship of the change of the high hover height (h_{min}) with change in the square of the critical speed (V_{cr}) $= \dfrac{d\ h_{min}}{dV^2_{cr}}$.

$\underline{f}$: Equivalent flat plate area - ft^2.

$\underline{A}$: Main rotor area - ft^2.

$\underline{I_R}$: Inertia of the dynamic system - slug ft^2.

$\underline{V_n}$: Any arbitrarily selected velocity between zero and the value of the critical velocity (V_{cr}) (used with Fig. 3 only)

$\underline{h_{n_1}}$: The height above the ground on the H-V diagram lower boundary that corresponds with the value of V_n (used with Fig. 3 only)

$\underline{h_{n_2}}$: The height above the ground on the H-V diagram upper boundary that corresponds with the value of V_n (used with Fig. 3 only)