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RESEARCH MEMORANDUM

INVESTIGATION OF THE STRUCTURAL DAMPING

OF A FULL-SCALE AIRPLANE WING

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NATIONAL ADVISORY COMMITTEE
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INVESTIGATION OF THE STRUCTURAL DAMPING OF A FULL-SCALE AIRPLANE WING

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SUMMARY

An investigation to determine the structural damping characteristics of a full-scale airplane wing was conducted by the shock-excitation method wherein the wing was loaded to a predetermined deflection and the load suddenly released. The test specimen vibrated at its fundamental bending frequency of 1.69 cycles per second. Only the first 2 or 3 cycles showed any indication of a higher frequency being superimposed upon the fundamental bending frequency. The damping was found to increase from about 0.002 of critical at an amplitude of vibration of ±0.05 inch to approximately 0.006 of critical at an amplitude of ±5 inches.

INTRODUCTION

The trend toward larger and faster aircraft has placed increasing emphasis on the importance of the dynamic response properties of airplane wing structures. One of the parameters involved in the computations of these dynamic response characteristics is the structural damping factor. Although some experimental data are available concerning the damping properties of full-scale airplane wing structures at relatively small amplitudes of vibration, very little data are available at large amplitudes. In connection with one phase of a fatigue program on full-scale airplane wing structures, it became necessary to determine the damping characteristics of the structure being tested. This paper presents the results of that test and in addition shows the effect of amplitude of vibration on the damping factor.

DESCRIPTION OF APPARATUS

The investigation described herein was conducted on a modified wing of a C-46D airplane which had been subjected to about 600 hours of flight service. The dimensions of the unmodified wing are as given in table 1.

The structural elements of the test specimen were typical of modern airplane wing structures, being of the riveted, stressed skin, two-spar construction with conventional ribs and hat section stiffeners. The spars were made up of relatively heavy T-shaped extrusions for flanges and sheet material reinforced by extruded angles for shear webs.

Because the primary purpose of this test specimen was for use in a fatigue program on full-scale airplane wing structures, certain modifications were made to the airplane. The original wing-fuselage attachments were left intact by cutting the fuselage in front of and behind the wing. The section of the fuselage incorporating the wing was inverted and supported between structural steel backstops. The left and right wing outer panels were cut off at the station 405 inches from the center line of the wing-fuselage combination; thus the span was reduced from 1296 inches to 810 inches. The shear material of the two spars in both the left and right wing panels, from approximately station 305 to station 405, was substantially increased. This modification was necessary to accommodate concentrated masses, or weight boxes, which were used in the fatigue tests to reproduce level-flight stresses at station 214. The centers of the masses were located at station 414 on both the left and right semispan. Various other local modifications also were made at station 405 to accommodate the concentrated masses. It was thought, however, that the modifications necessary to support the weight boxes would not materially affect the damping characteristics of the test specimen. A general view of half the test setup, which was symmetrical about the center line of the wing-fuselage combination, is shown in figure 1. The figure shows the fuselage supported between the structural steel backstops and the left wing with weight box attached. Also may be seen the release mechanism attached to the box and a flexible cable running to a hydraulic ram. This ram was the load actuator used to obtain the required values of initial deflection. The release mechanism was a triggered toggle joint actuated by rupturing a bolt with an explosive charge. The explosive caps in the two release mechanisms used, one on each semispan, were wired in series and fired from a common switch. Releases were made simultaneously within 0.001 second.

The instrumentation consisted of four acceleration-sensitive pickups and a recording oscillograph. Three of the accelerometers were located on the left weight box at station 414 with one each on the leading edge, the 30-percent-chord position, and the trailing edge. The fourth accelerometer was located on the right weight box at the 30-percent-chord position of station 414. The output from the acceleration-sensitive pickups was fed through the necessary balance boxes into a recording oscillograph, and simultaneous time histories of the vibration were obtained.

TEST AND PROCEDURE

The tests consisted of shock-exciting the wing specimen and then measuring its decay function as the resulting amplitude of vibration decreased to zero. The wing was shock-excited by loading each semispan symmetrically until a predetermined deflection was reached and then simultaneously releasing the applied loads by the quick-release mechanisms. The loads were applied by the hydraulic rams through the flexible cable and quick-release mechanism. The concentrated loads thus applied acted through the center of mass of the weight boxes. This center of mass was located at the estimated center of pressure of the air load outboard of station 214, if an air-load distribution corresponding to a positive low angle-of-attack condition is assumed.

Tests were made with six initial incremental deflections (measured at station 414) ranging from 1.6 to 5.6 inches. In each test the deflection was increased over each previous deflection by increments of 0.8 inch.

Immediately prior to releasing the load at each value of initial deflection, the recorder was turned on and a complete record of the acceleration was obtained as the wing vibrations decreased to very small amplitudes.

DISCUSSION AND RESULTS

A sample time-history record as measured by four accelerometers is shown in figure 2. The difference in the magnitude and direction of the various traces shown in this figure is not necessarily due to differences in absolute acceleration and direction since these values depend directly upon the calibration of each accelerometer. Check calibrations which were accomplished immediately following the present investigation indicated that the accelerometer used at the 30-percent-chord position on the left wing was slightly faulty; this defect accounts for the irregular shape of that particular trace. These calibrations also showed that all four accelerations were in phase. The smoothness of the traces shown in figure 2 indicates the lack of superimposed frequencies. Only the first 2 or 3 cycles of vibration, after the release had been made, showed any indication of higher frequencies being superimposed upon the fundamental wing bending frequency of 1.69 cycles per second. The fundamental bending frequency of the unmodified wing of the C-46D airplane, as found during ground vibration surveys, was 5.6 cycles per second. This reduction in frequency is due to the addition of the concentrated masses at station 405.

Data reduction for the present tests was accomplished by measuring the maximum amplitude for each cycle on the acceleration time-history records. Calculations then permitted the determination of the actual amplitude of vibration of the wing. These data were then plotted on semilog paper as a function of cycles of vibration. A composite of all the data is shown by the curve of figure 3. The spread in data from which this curve was determined was about ±3 percent. Figure 3 is significant in that the line faired through the test points is a curve rather than a straight line. This indicates that either the damping factor is not constant or the damping is not viscous.

On the assumption that viscous damping existed, computations for the damping factor were then made by utilizing the curve of figure 3 and the following equation:

$$\frac{C}{C_c} = \frac{1}{2\pi N} \log_e \frac{X_n}{X_{n+N}} \tag{1}$$

where

C viscous damping coefficient, pound-seconds per inch

C_c critical damping coefficient, pound-seconds per inch

$\frac{C}{C_c}$ damping factor

$N = 60$

$\frac{X_n}{X_{n+N}}$ ratio of amplitudes, 60 cycles apart, taken from any line tangent to the decay curve

Equation (1) is a modified form of the basic logarithmic decrement equation which can be found in references 1 and 2. A large value of N was chosen ($N = 60$) to reduce the inherent error in taking small differences of relatively large numbers when the reading accuracy is a fixed quantity.

Since in flutter calculations the parameter g is generally more convenient to use than C/C_c, it might be well to point out here that $\frac{C}{C_c} = \frac{g}{2}$. This relationship has been derived in reference 3.

The results of the computations to determine the damping factor are shown in figure 4 where amplitude of vibration is plotted as a function of the damping factor, C/C_c, on rectangular coordinate paper. This figure shows that the damping factor increases from about $\frac{C}{C_c} = 0.002$ for an amplitude of vibration of ±0.05 inch to about 0.006 at an amplitude of ±5 inches. No values of damping factor were obtained at higher amplitudes than those shown because of the strength limitations of the test specimen. Similarly, no lower values of damping factor were obtained because of instrument limitation.

Figure 4 also shows the damping factor plotted as a function of total maximum wing deflection and percent of estimated ultimate wing deflection. The mean deflection, that is, the deflection resulting from the concentrated masses, is 3.2 inches. The amplitudes of vibration start about this value of mean deflection which is 21.6 percent of estimated ultimate deflection. Thus, although the maximum amplitude of vibration measured by the accelerometers is only about ±5 inches, the total maximum deflection at the point is about 8.2 inches or 55.4 percent of the ultimate deflection. The assumption seems reasonable, however, that the value of mean deflection would not materially affect the results shown in figure 4 unless it were large enough to cause the maximum amplitude of vibration to exceed the yield strength of the wing. It also seems reasonable to assume that the curve of figure 4 does not approach a value of $\frac{C}{C_c} = 0.0065$ asymptotically as is indicated by the curve, but rather that, as the ultimate deflection is approached and the yield strength of the material is exceeded, the curve would have a point of inflection and then approach some horizontal line asymptotically. More tests would be necessary to substantiate this assumption.

The values of damping factor presented herein are assumed to be parameters of structural damping only. This assumption is substantiated by the results presented in reference 4 which show that tests on identical wing panels at pressures of 1 inch of mercury and 30 inches of mercury revealed no apparent change in damping factor.

The information in reference 4 also indicated a sizable frequency effect on the damping factor, but the author points out that the values obtained can be regarded as only of the correct order of magnitude with no claim to accuracy. The data of reference 4 do indicate, however, that the damping is doubled when the frequency is increased by a factor of about 3.5. (The factor of 3.5 is also the approximate ratio of the unloaded C-46 wing bending frequency to the frequency of the present test specimen.) It is interesting to note that when this ratio of frequency to damping is applied to the data presented herein, reasonable agreement is realized with the data of reference 4. Until more comprehensive tests are made, however, the values of damping factor presented herein are applicable only at a frequency of 1.69 cycles per second and caution must be exercised in applying the data to structures with higher frequencies.

CONCLUDING REMARKS

An investigation to determine the structural damping characteristics of a full-scale airplane wing was conducted by the shock-excitation method wherein the wing was loaded to a predetermined deflection and the load suddenly released. The test specimen vibrated at its fundamental bending frequency of 1.69 cycles per second. Only the first 2 or 3 cycles showed any indication of a higher frequency being superimposed upon the fundamental bending frequency. The damping was found to increase from about 0.002 of critical at an amplitude of vibration of ±0.05 inch to approximately 0.006 of critical at an amplitude of ±5 inches.

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REFERENCES

1. Den Hartog, J. P.: Mechanical Vibrations. Second ed., McGraw-Hill Book Co., Inc., 1940.

2. Myklestad, N. O.: Vibration Analysis. McGraw-Hill Book Co., Inc., 1944.

3. Coleman, Robert P.: Damping Formulas and Experimental Values of Damping in Flutter Models. NACA TN 751, 1940.

4. Theodorsen, Theodore: Miscellaneous Tests in Vacuum Sphere on Structural Damping of Airplane Wing Panels. NACA RB, Oct. 1942.

TABLE 1

GEOMETRIC CHARACTERISTICS OF C-46D WINGS

Airfoil section from center line to station 192	NACA 23017
Airfoil section from station 192 to tip changes gradually to	NACA 4401.5
Wing area, square feet	1360
Wing span, feet	108
Tip chord, theoretical, inches	66
Root chord, inches	198
Taper ratio	0.333
Mean aerodynamic chord, inches	164.25
Incidence at wing root, degrees	3.5
Incidence at wing tip, degrees	0.5
Dihedral at 70 percent chord, degrees	7.0
Sweepback, leading edge, degrees	11.4
Aspect ratio	8.58

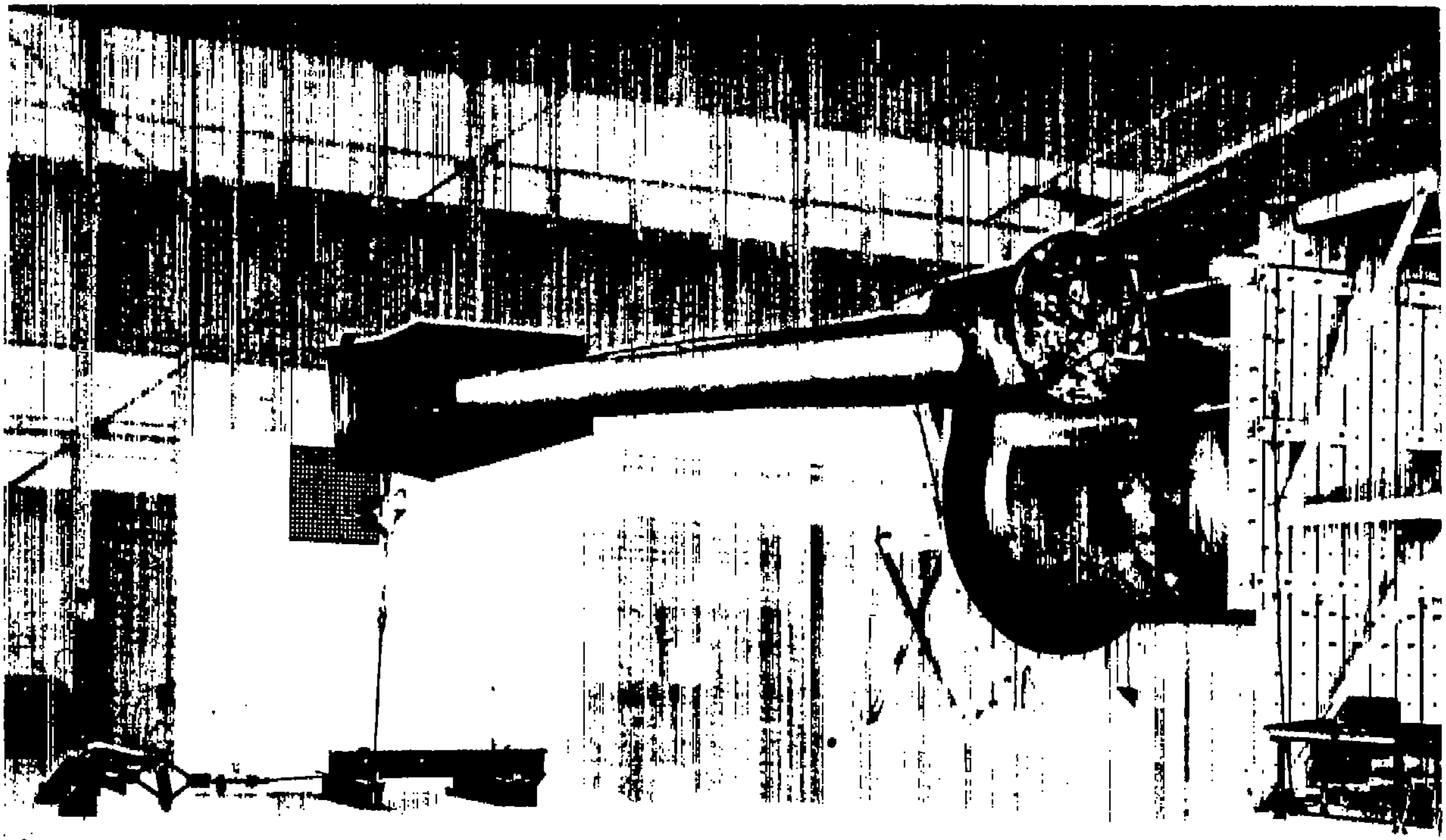

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Figure 1.- General view of left half of test setup.

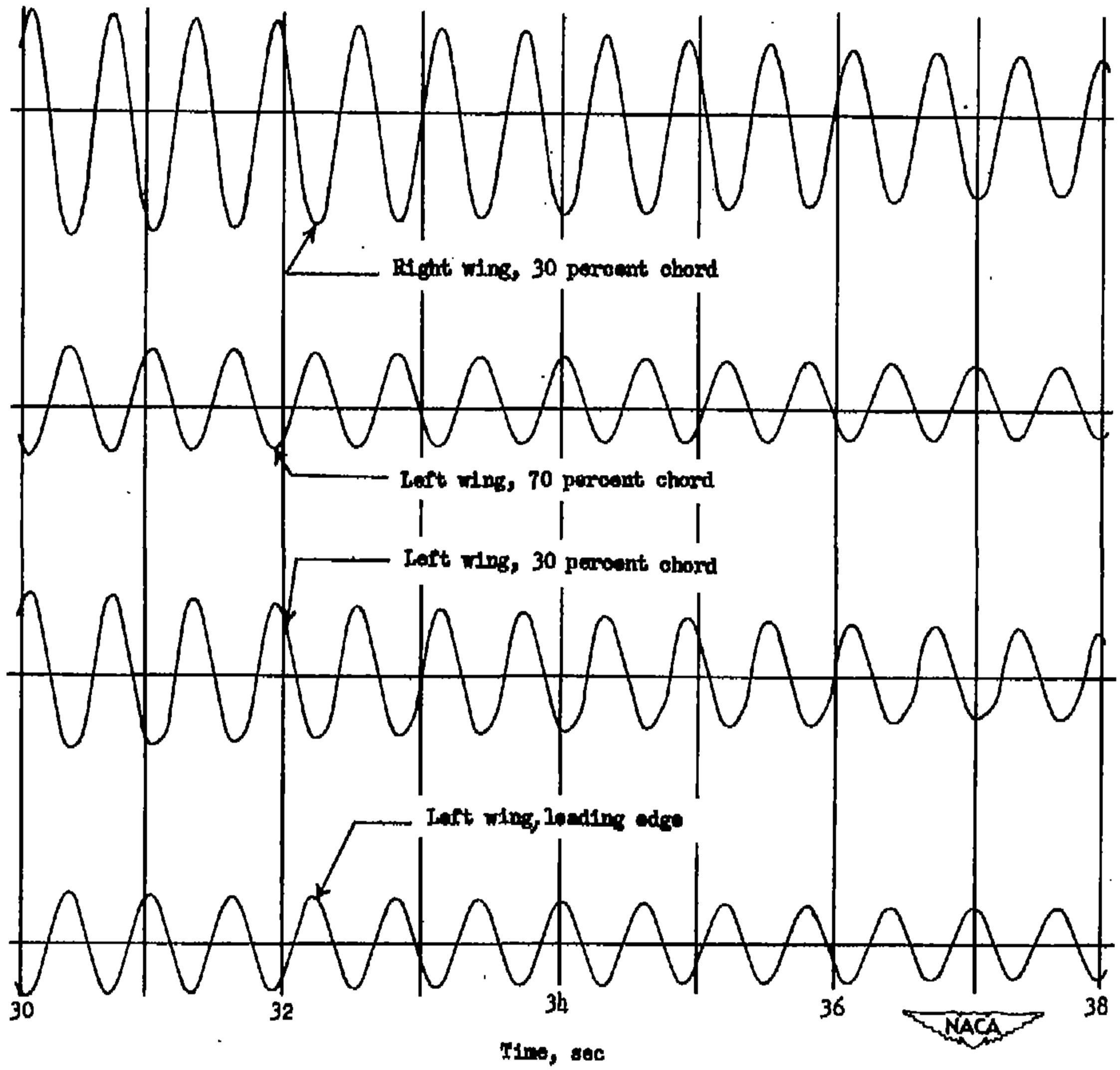

Figure 2.- Sample of vibration time-history record for four accelerometers.

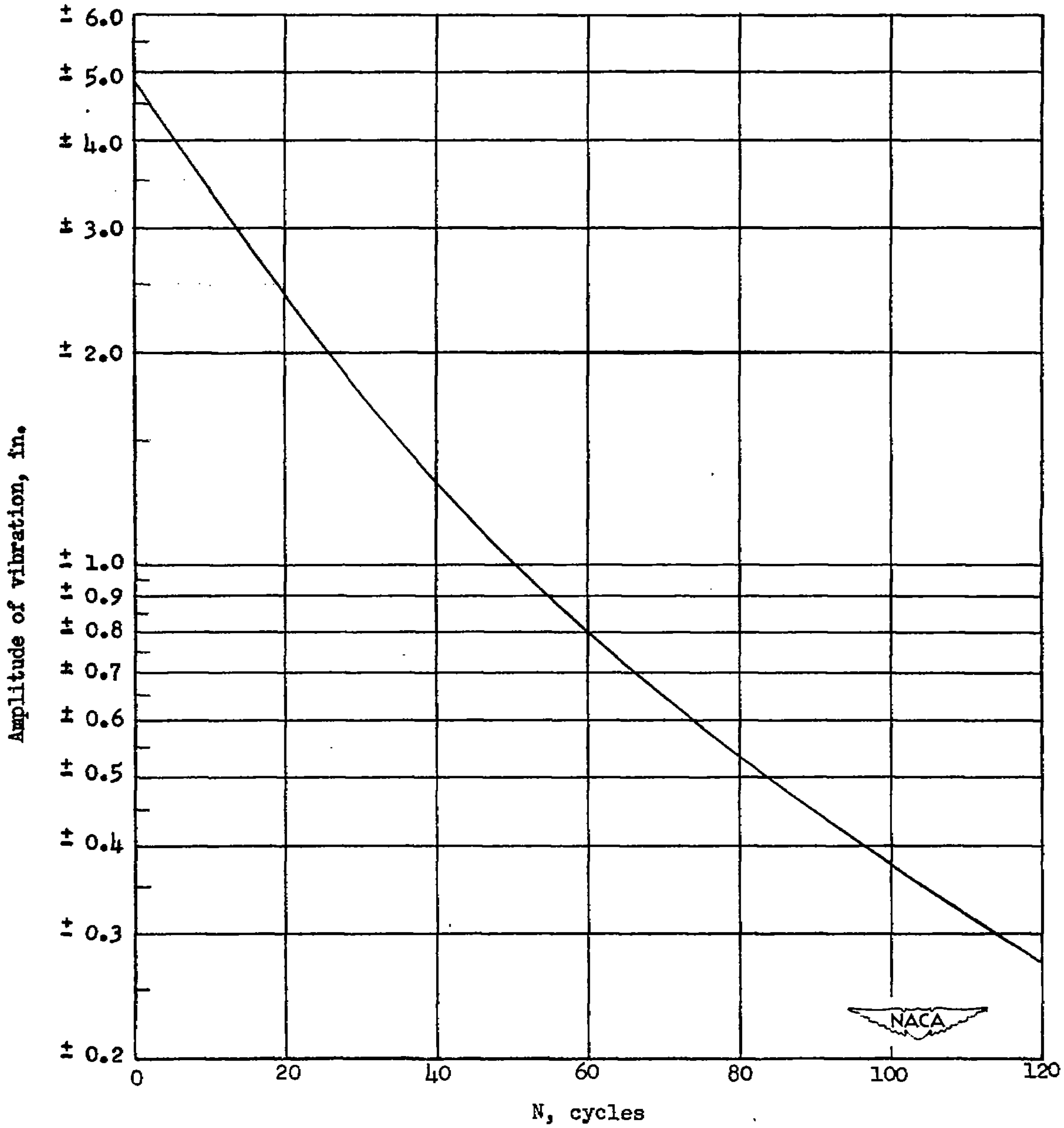

Figure 3.- Decay function of wing for combined data.

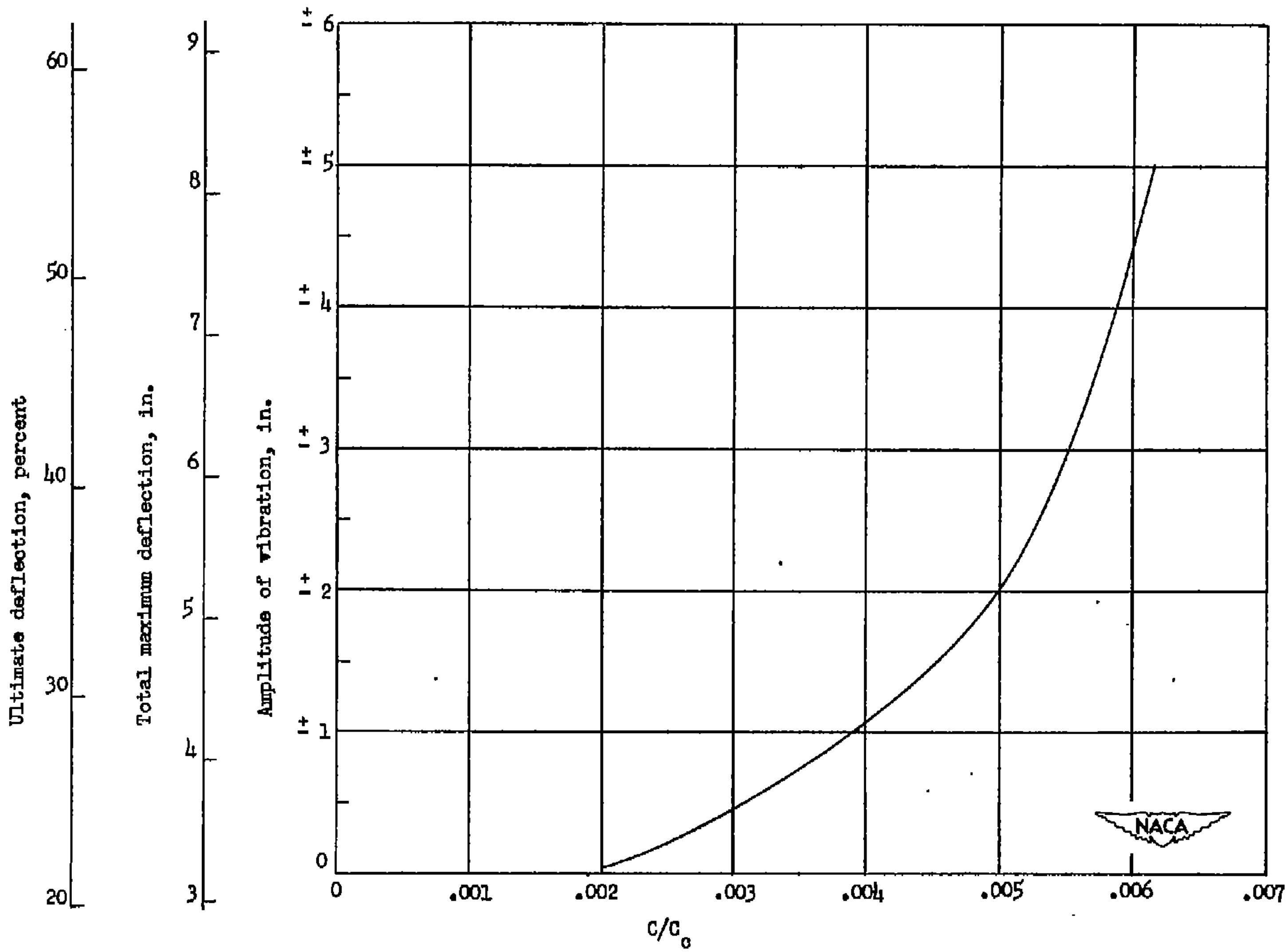

Figure 4.- Variation of damping factor with amplitude of vibration.

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