EFFECTS OF AEROELASTICITY ON THE STABILITY AND CONTROL CHARACTERISTICS OF AIRPLANES

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INTRODUCTION

Aeroelasticity has had a direct influence on aircraft design from the earliest days of flight. For instance, the Wright Brothers utilized the low torsional stiffness of the outer wing panels by actually distorting the panels by a system of wires to provide lateral control. Also, it is now felt that Langley's attempt at flying failed due to aeroelastic distor-
L tion of the lifting surface, which resulted in wing divergence. Although
1 World War II aircraft had their share of aeroelastic problems, the genera-
4 tion of jet aircraft after the war, having thinner wings and higher per-
4 formance, developed a multitude of aeroelastic problems. These included
6 flutter, loss of control, control reversal, divergence, buzz, etc. It is reasonable to extrapolate to the next generation involving the supersonic bomber, and transport, that these problems will be multiplied, particularly in view of the deleterious effect of aerodynamic heating.

The particular aspect of aeroelasticity to be considered in this report, i.e., effects on stability and control, is a broad subject and all facets cannot be discussed. Therefore, some recent stability and control problems and some methods used to correct them will be touched on. An attempt will be made to indicate some problems which might be expected to accompany near-future designs.

The paper is divided into essentially four sections; first, a brief overall look at the flight dynamic field will be presented; second, a discussion of aeroelastic stability and control problems which have occurred on recent aircraft with some specific examples; third, a brief look at some of the basic inputs including elasticity and aerodynamics; and finally, some recent work at Langley Research Center will be discussed including the effects of aerodynamic heating, wing deformation, and the effect of fuselage elasticity on stability and control of a large supersonic airplane.

The present subject has been treated by a large number of contributors, and it is, therefore, impracticable to give individual credit. A representative bibliography, however, can be found in references 1 and 2. Most of the information was obtained from the most accessible sources and therefore is weighted heavily with U.S. experience.

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ORIENTATION OF PROBLEM

This first section of the paper will be concerned with a discussion of the general flight dynamic picture with particular reference to the stability and control aspects; especially, the relationships of the problem of stability and control of the flexible airplane to that of the rigid airplane and also to other aeroelastic problems. These relationships are easily visualized by referring to the well-known triangle of forces originated by A. R. Collar of Great Britain. This triangle is shown in figure 1. The three primary forces indicated are inertial, aerodynamic, and elastic. As shown in the center box, all three forces are involved in the subject of stability and control as well as in the subjects of flutter of the flexible aircraft, and structural feedback involving a coupling of the system elasticity with the automatic control systems. The subjects of stability and control and the subject of flutter may be distinguished in that stability and control usually apply to the aircraft as a whole, whereas flutter is often associated with components of the aircraft such as wing, tail, and control surfaces.

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Other aspects of aircraft dynamics are indicated on the outside of the triangle. The box on the right side refers to the dynamics of the rigid airplane, a well-documented subject. The subject of the stability and control of the flexible aircraft may be regarded as a generalization or as an extension of that of the rigid aircraft. The basic definitions and criteria developed for rigid-body analysis have been carried over into the flexible body analysis. Of course, certain additional quantities are required in the flexible body analysis to account for static and dynamic coupling.

Consider now the box on the left side of the triangle. This contains, as examples, structural divergence, and control effectiveness and reversal. These phenomena usually involve only the aerodynamic and elastic forces on components of the aircraft.

It is perhaps correct to say that almost all past stability and control problems have been treatable by the subjects in the left and right boxes rather than by the general subject indicated in the center box. That is, it has been sufficient to utilize essentially rigid-body analysis together with the associated derivatives modified to include static effects of structural deformation. Thus, inertial and velocity effects of structural deformation which are a part of the general subject (center box) have been considered to be negligible and with justification for most problems. This approach is commonly referred to in the literature as the quasi-static method. Near-future configurations (supersonic transport), however, may require a general analysis indicated by the center box. In this manner, all aeroelastic problems including flutter and airplane stability would be treated simultaneously.

The general analysis also provides for the calculation of dynamic motions and loads responses to disturbances such as random gusts, as well as control motions. With regard to random gusts, although the subject of stability and control traditionally has been treated separately from the subject of loads on airplane structures, there is a growing appreciation of the rather close relationship between the magnitude of random gust loads and the dynamic stability characteristics of the aircraft. This relation is significant for designs for which the damping ratios of the short-period or dutch roll modes are small (say less than one-tenth of critical damping). In such cases the level of gust loads increases appreciably as the damping ratio is reduced. Some effects of short-period longitudinal stability characteristics on airplane response to random turbulence are presented in reference 3.

A new element has recently entered the picture, namely the deteriorating effect of aerodynamic heating on the structural capability. The achievement of supersonic flight speeds has introduced a complicating factor by causing elastic properties of materials to change with operating conditions due to thermal effects. For missiles, elastic forces may even become time dependent, and this point will be discussed later in the paper.

Another factor that can enter the picture for the supersonic aircraft is fuel slosh. For large missiles having perhaps 90 percent of its weight in liquid fuel, it is mandatory to consider the dynamics of the fuel. For large supersonic aircraft the percentage of fuel with respect to the total weight is rapidly increasing and this factor will probably have to be treated, including a coupling with the vibration modes.

SOME RECENT PROBLEMS

This section is concerned with some problems which have been encountered on operating aircraft.

Roll Characteristics and Aileron Control

One of the most common of aeroelastic problems which continues to plague the industry is the loss of effectiveness and reversal of aileron control. The phenomenon stems from torsional deformation of both straight and swept wings and from bending deformation of swept wings. These effects have been experienced in the design of sweptwing jet fighters and more recently in the design of jet transports. An example of loss of aileron effectiveness and reversal is shown on figure 2. The solid line is the ratio of rolling helix angle of a sweptwing jet fighter to the angle for the airplane considered rigid $(pb/2V)_f/(\rho b/2V)_r$, and is plotted against Mach number at a constant altitude. Note aileron reversal at about Mach number 0.95. The dashed curve on the slide is the same ratio for a similar airplane modified to avoid reversal by shifting the ailerons inboard and by providing increased torsional stiffness of the wing structure. Note that the resulting aeroelastic effect is to increase somewhat the control effectiveness over that for the rigid wing. This illustrates that with proper design, the distortion of the wing under load may be advantageous. A similar fix was necessary in the case of at least one of the turbojet transports. Inboard ailerons were used; outboard ailerons were locked out at high speeds, and control was augmented by use of spoilers.

One of the techniques employed in the preceding cases was a redistribution of aerodynamic forces more favorable to a reduction in accumulated torque. Figure 3 illustrates the gain in aileron reversal speed obtained as a result of changing aileron configuration. In this case, however, the aileron remains outboard and its span is increased. Reversal speed, plotted on the vertical scale, is seen to increase with increasing aileron span. It is apparent that further gains can be obtained by shifting the ailerons generally inboard or by splitting them and locking out the outboard section at high speeds.

Flexible Tail Surfaces

Another aeroelastic problem involving control effectiveness is the flexibility of tail surfaces or of their attachment points. This flexibility has frequently caused a reduction of stability and of control effectiveness such as the magnitude of pitching velocity obtainable per unit elevator deflection. A recent example was a reduction in elevator effectiveness due to stabilizer flexibility at one stage of the design of a jet transport airplane. Stabilizer flexibility did not appear to affect longitudinal stability in this case, however. The problem was resolved by increasing stabilizer stiffness.

A particular instance of marginal directional stability caused by bending deformation of a swept vertical fin under load has also been reported. As indicated on figure 4(a), the bending deformation of a swept surface under airload reduces the local angles of attack (as illustrated at the top) relative to the root angle of attack. The force developed per unit root angle of attack is, therefore, smaller than that for a rigid surface. Attempts to improve stability by increasing the bending stiffness led to a reduction in flutter speed due to the approach of the ratio of bending to torsion frequencies to a value of unity, an undesirable condition from a flutter standpoint. Flutter speed was then raised by adding torsional stiffness. The torsional deformation, however, was such as to improve directional stability, as indicated on figure 4(b). Here, due to twist about the elastic axis, the local angle of attack is greater than the angle at the root; hence, the force per root angle of attack is increased over that for a rigid surface. Increasing torsional stiffness, therefore, led again to poor directional stability. The cycle of stiffenings was repeated three times before both directional stability and flutter speeds were satisfactory. In this case the torsional stiffness requirements for the two aeroelastic effects were conflicting. It is indicated that arbitrary stiffening of the structure alone is not always the proper approach. One must consider the use of stiffening and favorable load redistribution and at the same time watch out for adverse changes in aeroelastic effects on other than stability and control.

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Effect on Automatic Control

Another class of problems concerns effects of flexibility of the aircraft structure on dynamic stability of an automatically controlled airplane. This phenomenon, often referred to as "structural feedback," is common in large missile designs and has been found to be of importance for several large aircraft. The dynamic instability is caused by spurious responses due to local structural deformations produced by an attitude or rate sensor which is supposed to respond only to the motion of the vehicle as a whole. An illustration of the origin of the spurious response is shown on figure 5. Here the first fuselage mode is shown with the rigid-body pitching motion indicated by θ. If an attitude sensor is placed at the indicated point, it will provide a signal composed of θ which it is supposed to measure and also γ the slope of the elastic curve. The pitch signal θ would normally call for a trailing edge down elevator deflection to provide the proper restoring moment. This elevator deflection is augmented by the angle γ and is in a sense as to increase the mode deflection, hence increase γ. If the sensitivity of the automatic control signal is sufficient, an oscillatory instability is likely to develop. The problem becomes of particular interest with regard to airplane stability if the modal instability develops for a sensitivity which is insufficient to stabilize the airplane as a whole. Although the first mode was shown in the illustration, higher modes could contribute to the problem. The problem can also occur in lateral motions.

Here the aeroelastic problem has been complicated by the coupling of forces by the automatic control system. Although methods of coping with this problem include techniques such as stiffness increases and air load redistribution, it is usually profitable to first analyze the effects of the position of the sensor on the structure and employ electrical filter circuits in the control system.

METHODS OF CALCULATION

Aeroelastic studies like others are pursued partly by experimental testing and partly by calculation. The general method of calculation is, of course, based on equations of motion including the inertial, aerodynamic, and elastic forces. The form of the equations is subject to variation depending upon the method used to express the elastic forces. The degree of complexity of the analysis depends upon the vehicle configuration parameters and operating conditions. The basic ingredients involving both elastic and aerodynamic forces are treated in a new AGARD Manual on Aeroelasticity edited by W. P. Jones which is now in final stages of preparation.

Elastic Forces

Expressions for the elastic forces may be developed from structural deformations described by structural influence coefficients or by a series of mathematical functions usually possessing orthogonal properties, for example, the free-free normal modes of the structure. The first method is often called the lumped parameter method, the second is called a modal method. Frequently the two procedures are combined by calculating the modes from a set of a large number of lumped parameter equations.

For structures for which simple beams are an adequate approximation, modes may be calculated in iteration procedures, or by assuming a series of natural modes of a uniform beam. For plate-like structures, structural influence coefficients are perhaps most commonly used. Several contributors to this subject are Levy (ref. 4), Schuerch (ref. 5), and Turner, et al (ref. 6).

Aerodynamic Forces

The aerodynamic input is usually obtained from a combination of experimental work and theoretical procedures. Certainly, experimental determination of stability derivatives has been, and is, one of the most fruitful sources of data; however, for the case of aeroelasticity where distortion of the wing results in a redistribution of the load, the scaled construction, instrumentation, and testing of a twisted and bent wing are a major and expensive undertaking and resort to theoretical procedures is desirable. In cases where a flutter model of the complete aircraft has been constructed, static and dynamic stability characteristics can be determined.

Several aerodynamic techniques which have been developed for use in other structural dynamic fields such as flutter may be applied to stability and control problems. Two such approaches have been termed the "kernel function" for use at subsonic and low supersonic speeds and "piston theory" applicable at high supersonic speeds. A brief discussion of these two procedures is given below.

Kernel function.- The kernel function method has been developed for determining the air force on a wing of arbitrary plan form which is oscillating in an arbitrary mode in subsonic and supersonic flow, and consists basically of numerical integration of the integral equation which relates the downwash or vertical velocity at a point on the wing to the loading on the wing. This integral equation takes the form

$$w(x,y) = \iint_{\text{Surface}} L(\xi,\eta)K(x - \xi)(y - \eta)dy\,dx$$

where

w vertical downwash which may be composed of angle of attack, wing motion, wing deformation

$L(\xi,\eta)$ unknown loading

$K(x - \xi)(y - \eta)$ kernel of the integral equation

The kernel K is a rather complicated function and is treated separately in reference 7, whereas the actual application for the subsonic case is treated in references 8 and 9. The supersonic case for subsonic leading edges has been accomplished and is in unpublished form at the Langley Research Center. Thus, for a particular wing angle of attack, motion, or deformation, the corresponding loading can be determined. For practical application the use of computing machines having a large capacity is mandatory.

As a comparison of the kernel function with an analytical procedure (ref. 10), the following table gives the lift and the location of the center of pressure for $M = 0$ for a circular wing which has a parabolic deformation in the chord direction:

Source	$C_{x_{x^2}}$	$C_{m_{x^2}}$	Center of pressure (percent from L.E.)
Analytical (ref. 9)	0.9436	-0.4382	26.80
Kernel function	.9443	-.4463	26.40

These results are within two percent of the analytical procedure.

A comparison of the results of kernel function calculations with experimental results for a rectangular wing oscillating about a roll axis can be made from figure 6. The results are in the forms of lift coefficient and associated phase angle as functions of a frequency parameter, $k = \omega c/2V$, where ω is the frequency of oscillation, c is the chord, and V the forward velocity. On the upper portion of the slide, excellent agreement is shown between the experiment and theory for the amplitude ratio, whereas on the lower portion of the slide fair agreement is obtained between the experiment and theory for the phase angle.

Piston theory.- Another aerodynamic concept which has been found useful for aeroelastic problems, particularly for flutter, is "piston theory." This method was used for calculating the flutter speed of the lifting surfaces of high-speed research airplanes in the high supersonic range and has shown excellent agreement with model tests. In addition, good agreement between piston theory results and experiments has been found on the flutter of highly swept delta wings at high Mach number.

Basically, a major assumption is that a vertical slab of air as it strikes a surface at high Mach number remains a slab and that the airfoil surface acts as a piston, operating in a direction normal to the flow. Therefore, the simple formula for the pressure on a piston is applicable. Lighthill, reference 11, presented this theory in a Journal of the Aeronautical Sciences several years ago. Additional applications are given in reference 12.

The advantage of this formulation is that it provides a point solution which is readily adaptable to a surface having a complicated structural distortion pattern, and in addition will provide solutions involving effects of airfoil thickness. At high speeds $(M > 3)$, and particularly for airfoil shapes having cross sections with negative slope such as a diamond airfoil, the center of pressure will not always be at the location predicted by flat-plate theory, and thus the moment on the wing may not be correct. Later in the paper, an illustration of the longitudinal stability of a supersonic aircraft will be discussed in which piston theory was used for the aerodynamic input.

SOME EXAMPLES OF RECENT NASA STUDIES PERTINENT TO THE STABILITY AND CONTROL OF FLEXIBLE AIRCRAFT

In this final section three NASA aeroelastic studies which are pertinent to contemporary and near-future aircraft configurations are presented.

Delta Wing

The first study is a comparison of the measured and calculated static deformations of and forces on a 45° delta wing model in the Mach number range of 1.3 to 4.0. The semispan of the model is six inches and the structure is solid steel. The airfoil section of the wing is a symmetrical double wedge having a thickness ratio of two percent. The wing is clamped at the root. Structural deformation calculations are based on measured structural influence coefficients. Aerodynamic forces were calculated by the use of an unpublished lifting-surface theory, which not only accounts for the bending deflection of the wing, but can treat the cambered deformation as well.

A typical plot of the experimental and calculated deformations under airloads is shown in figure 7(a) on which the deformations for several constant percent chord stations are plotted as functions of the distance from the root in percent root chord. The circles indicate the test results; the lines indicate the calculated results. The agreement is satisfactory. The chordwise variation of the deformation is shown in figure 7(b). Here deformation is plotted as a function of the distance along a local chord in percent of the local chord.

Some corresponding experimental and calculated aerodynamic forces are presented in figure 8 as functions of dynamic pressure for several Mach numbers. The normal force coefficient per unit angle of attack is shown on the left side and the pitching-moment coefficient per unit angle of attack is plotted on the right. As before, the test points are indicated by symbols and the calculated data by the solid lines. The dashed lines represent the behavior of a rigid wing. The test and calculated data show the same trends with dynamic pressure and Mach number and the agreement between the sets of data is fair. A comparison of the results for the flexible and rigid wings indicates that aeroelastic effects for this case have appreciably reduced the aerodynamic forces at large values of dynamic pressure for the lower Mach numbers. Were this lifting surface to be used as a stabilizer or control, the degree of stability and control would be lessened by the deformation of the structure.

Thermal Effects

The next study was chosen to illustrate a pertinent aeroelastic effect brought about by high-speed flight. Aerodynamic heating which can influence the stiffness of a system is a newer element in the stability problem. There are two separate conditions which can influence the system stiffness; one is a long time soak at an elevated temperature which manifests itself principally in a loss in both bending and torsional stiffness due to the change in material properties. Another effect of aerodynamic heating manifests itself in highly accelerated flight, and very large losses in torsional stiffness can occur over a short time interval. This is particularly true for solid surfaces such as might be found on a small missile. This effect is due to the fact that the trailing and leading edges of the wing, having much less material than the center section, are heated to higher temperatures and expanded, which results in a reduction in torsional stiffness. As an illustration of the loss in torsional stiffness, figure 9 presents the calculated variation in torsional stiffness at one spanwise station at various times for a solid aluminum rectangular wing which, in a heated wind-tunnel test, was suddenly subjected to $M = 2$ flow having a stagnation temperature of 800° F. Note that the stiffness decreases and then increases with time as the temperature distribution becomes more uniform across the chord. The wing actually fluttered for two seconds and then ceased, because the torsional stiffness increased again. However, here it is being used as an example of how the wing might behave under load at the various times of flight, if flutter were suppressed by say a change in mass distribution.

The particular quantity used to illustrate the effect of aeroelasticity under transient aerodynamic heating is the lift-curve slope C_{L_α}. This quantity was calculated for various times using piston theory aerodynamics and appropriate values of GJ exemplified by the previous slide. Numerical integration was used in an iterative scheme to obtain the deformations and values of C_{L_α}. The results are shown in figure 10. Here C_{L_α} for the heated flexible wing is divided by the C_{L_α} for the rigid wing and plotted as a function of time. Note that at zero time corresponding to a cold structure the ratio is greater than one. This indicates that the configuration has a divergent tendency; that is, the center of pressure is ahead of the elastic axis. Now at progressively greater elapsed times the ratio increases sharply by a large factor as a consequence of the reduction of torsional stiffness due to the temperature differential between the center and the leading and trailing edges. In the present example the loss of torsion is undoubtedly aggravated by the solid-wing structure. However, the problem is likely to be significant for built-up wing structures as well. It is concluded that for configurations subject to a large amount of rapid aerodynamic heating design for aeroelastic effects on the basis of a cold structure could lead to difficulties in flight.

Effect of Fuselage Flexibility on Supersonic Transport Airplane

The third example of aeroelastic effects is a brief theoretical study of the dynamic stability and static control characteristics of a supersonic transport configuration. As shown in figure 11, the configuration is a canard form with triangular lifting surfaces. Longitudinal control is obtained through movement of the canard surface.

In the study the fuselage and wing were treated together as a flexible longitudinal beam, the stiffness of which was progressively reduced for a trend study. The motion and deformation of the airplane were approximated by rigid-body normal and pitching motion together with the first three fuselage free-free bending modes. The fuselage modes are shown in figure 12. Note that wing deformation is restricted to chordwise bending. The airspeed is assumed constant. The equations of motion were derived by a Lagrangian formulation. The flexible modes were obtained by a numerical iterative procedure and piston theory was used to calculate the aerodynamic forces.

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The equations of motion were first used to obtain the characteristic eighth-order polynomial which was then factored into four quadratic factors. Each factor represents an aerodynamically coupled mode of motion. From each quadratic the undamped natural frequency and damping ratio were obtained to describe the stability of each mode; that is, static instability is indicated if a natural frequency becomes imaginary, and dynamic instability is indicated if the damping ratio becomes negative.

Dynamic stability.- The flight condition considered was the cruise condition at a Mach number of 3 at 60,000 feet. The results of this study are shown in figure 13. In figure 13(a) undamped natural frequencies of the various modes of motion have been divided by the undamped short-period frequency for the airplane considered rigid and are plotted as a function of λ, a stiffness parameter which is proportional to the stiffness of the fuselage. A rigid airplane is indicated by $\lambda = \infty$. Current design practices are represented by values of λ in the vicinity of 1.0. Future configurations may result in values of λ considerably less than 1.0

As λ is reduced the frequencies of the flexible modes drop more or less in proportion. The short-period frequency decreases very slightly. As λ is reduced to rather low levels the decrease in the frequencies of individual modes is no longer proportional. They actually cross; the short period even increases. It is of interest to note that no evidence of static instability appears on this figure.

The damping ratios of the modes of motion as functions of λ are presented in figure 13(b). At the higher values of λ the damping ratio of the short-period mode is much larger than those of the flexible modes with the second and third mode damping ratios being essentially the same. As λ is reduced the damping ratios of the flexible mode increase while that of the short period decreases. At very low values of fuselage stiffness the damping ratio of the second mode becomes negative indicating a dynamic instability. At a still lower value of stiffness the short period becomes dynamically unstable as indicated by its damping ratio.

Effect of analysis simplification.- The preceding results included three flexible fuselage modes. It is of interest to compare these results with those from a simpler analysis in which the fuselage deformation was described by only the first flexible mode. This can be done with figure 14 on which the curves for the short period and first fuselage modes from figure 13 are repreduced by the solid lines. Note that both the frequency ratios (at the bottom of fig. 14) and the damping ratios (at the top) are shown. The corresponding results of the simpler analysis which excludes the second and third modes are shown by the dashed lines.

Little difference in the results of the two analyses appears for values of λ greater than about 0.3. Slight differences in the frequencies occur at low values of λ. The differences in damping ratios at low values of stiffness are more marked. The short-period instability, however, as shown by the passage of the damping ratios through zero is indicated by both methods. The simpler method, of course, does not indicate the instability of the second mode shown on the previous figure.

Both of the methods mentioned are dynamic analyses represented by the box in the center of the triangle of forces shown on figure 1. It was mentioned earlier that the use of essentially rigid-body analysis modified by the forces due to static deformations (quasi-static analysis) was useful for most stability studies in the past. A comparison of results of dynamic and quasi-static methods for the present problem can be made with figure 15. The results of the dynamic analysis using three flexible fuselage modes for the short-period mode are again reproduced by the solid lines. The results of the quasi-static analysis, also using three flexible fuselage modes, are shown by the dashed lines.

Little difference in the results of the two methods is indicated for values of λ above about 0.4. In fact, for the short-period frequency only small differences are found over the entire range of stiffness.

A significant difference, however, in the damping ratios of the short-period mode is noted for values of λ less than about 0.4. The quasi-static method does not indicate the dynamic instability shown by the dynamic method.

The choice of whether to use the quasi-static or the dynamic analysis depends upon the ratio of aerodynamic to elastic forces. A rule of thumb which has been used and is substantiated by the present results is that quasi-static analysis is likely to be satisfactory if the lowest structural frequency is several times greater than the short-period frequency. As for all simple criteria, it should be used cautiously, particularly for unusual configurations and operating conditions. It is emphasized that the simple rule applies only to effects on longitudinal stability and not to dynamic instabilities in the nature of flutter.

There are no simple guides as to how many flexible modes are needed to provide a satisfactory dynamic analysis. This has been demonstrated in the history of flutter analyses.

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It should be borne in mind that the results presented here are to illustrate the possibility of stability problems and not to determine the likelihood of occurrence. Factors which influence the occurrence of dynamic instabilities include not only the fuselage stiffness and dynamic pressure which were considered here, but also such factors as the location on the canard surface and the distributions of mass and elastic properties.

Although the method applied here in general can be used to determine flutter instabilities as well as airplane instability, all likely flutter instabilities might not be present in the results shown. For example, the wing deformation allowed in this study was limited to chordwise bending and, therefore, instabilities involving spanwise wing deformations did not appear.

Control effectiveness.- Some information on the effect of the three fuselage modes on canard control effectiveness was obtained and is shown in figure 16. Here is shown the steady pitching velocity per unit canard deflection and the contribution by each of the three flexible modes to the total deflection plotted as a function of stiffness, λ. The curves are cut off at a value of λ just before the dynamic instability shown on the previous figure.

It can be seen that the control effectiveness increases somewhat as the stiffness is reduced. Also indicated is a sharp increase in the contribution of the second flexible fuselage mode to the total deformation at low stiffness levels.

CONCLUDING REMARKS

Some summarizing remarks can be made from the preceding selective coverage of the aeroelastic effects.

The most common aeroelastic effects on airplane stability and control appear to be associated with the deformation of wings and stabilizing surfaces. The most frequent occurrence appears to be loss of aileron effectiveness and reversal due to wing deformation. Some loss of directional stability and elevator effectiveness has been reported recently.

It is likely that these effects will continue to be present in future designs such as a supersonic transport airplane. It appears that the dynamic stability of the supersonic transport may also be adversely affected by fuselage flexibility.

Effects of aerodynamic heating may be severe, particularly for solid surfaces such as might be used on a small missile and estimates of the changes in the load distribution should be determined in practical cases.

Techniques of minimizing adverse elastic effects involve stiffening of the structure and also in rearranging the distribution of aerodynamic forces in a more favorable manner. These approaches, of course, must be used in such a manner as to avoid worsening other aeroelastic characteristics such as tendencies toward divergence and flutter.

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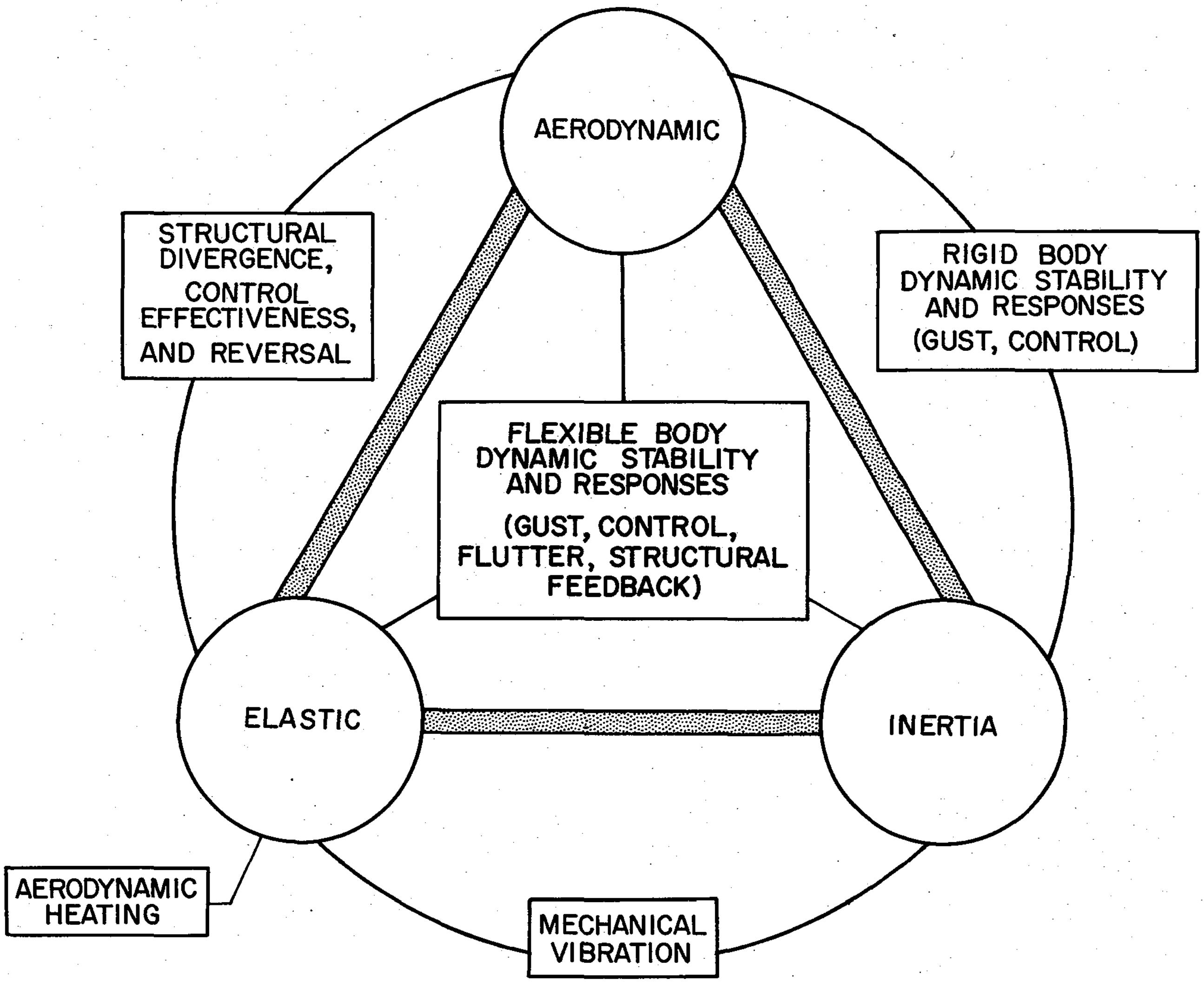

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Figure 1.- Triangle of forces.

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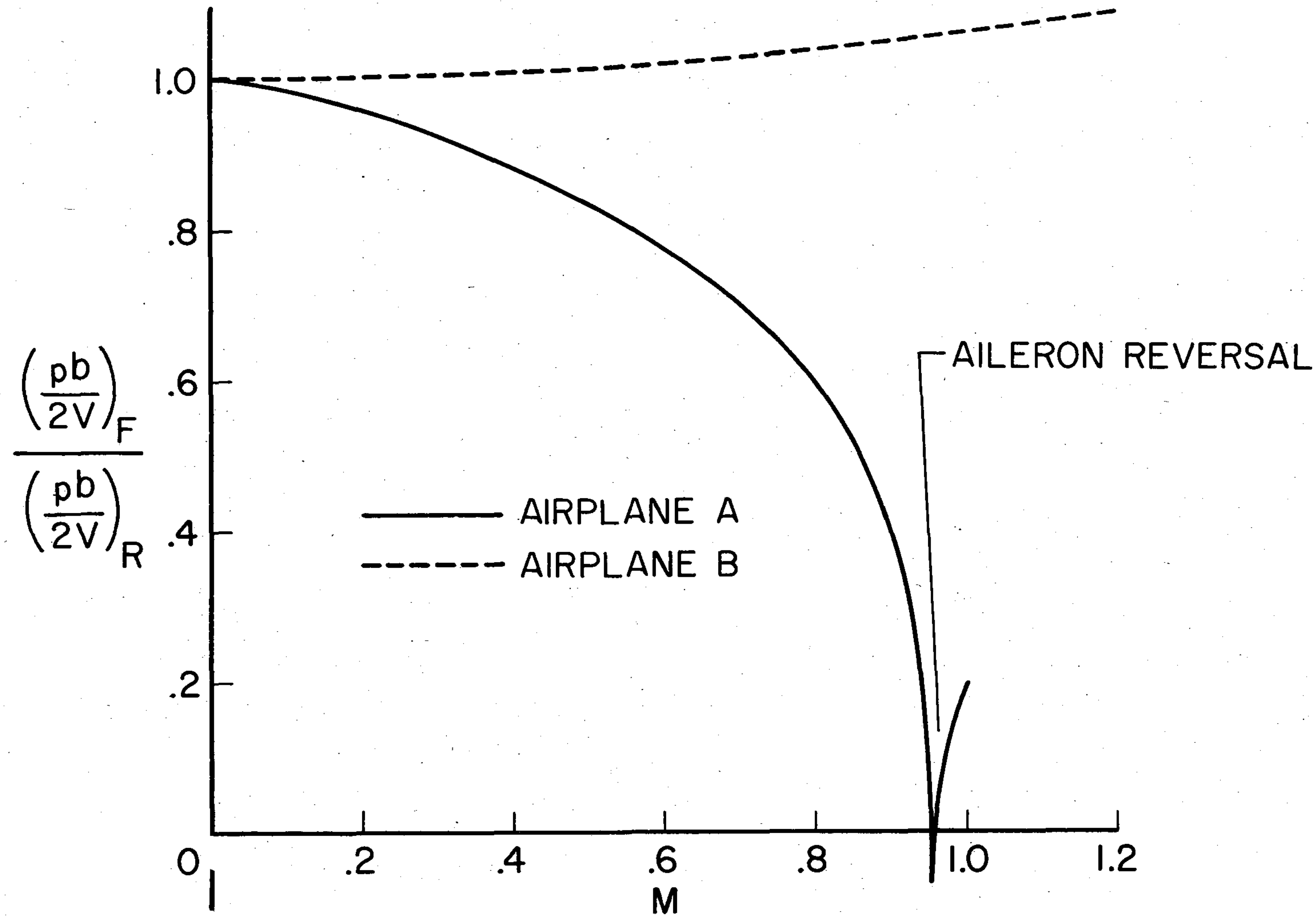

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Figure 2.- Influence of aeroelasticity on roll rate for fighter type aircraft.

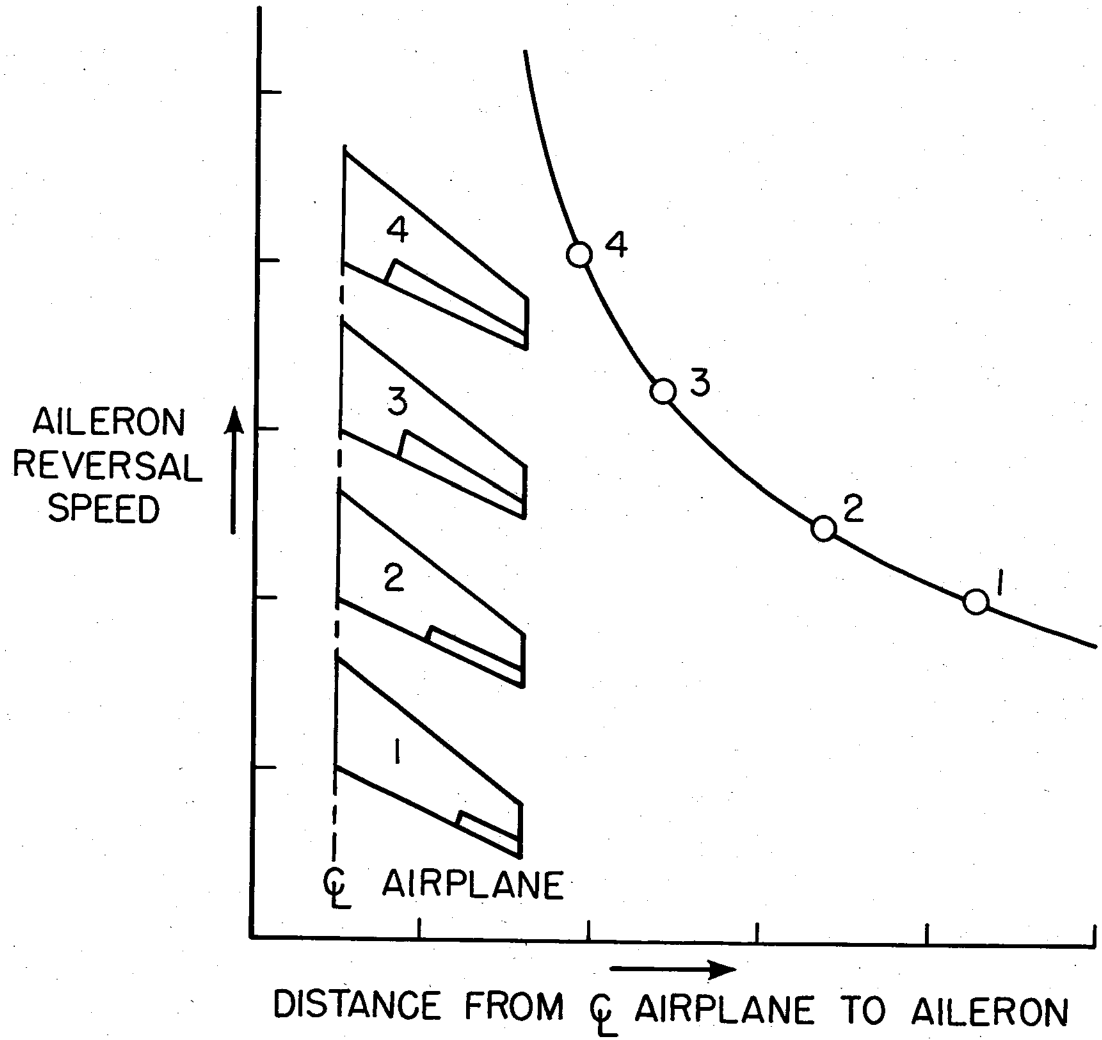

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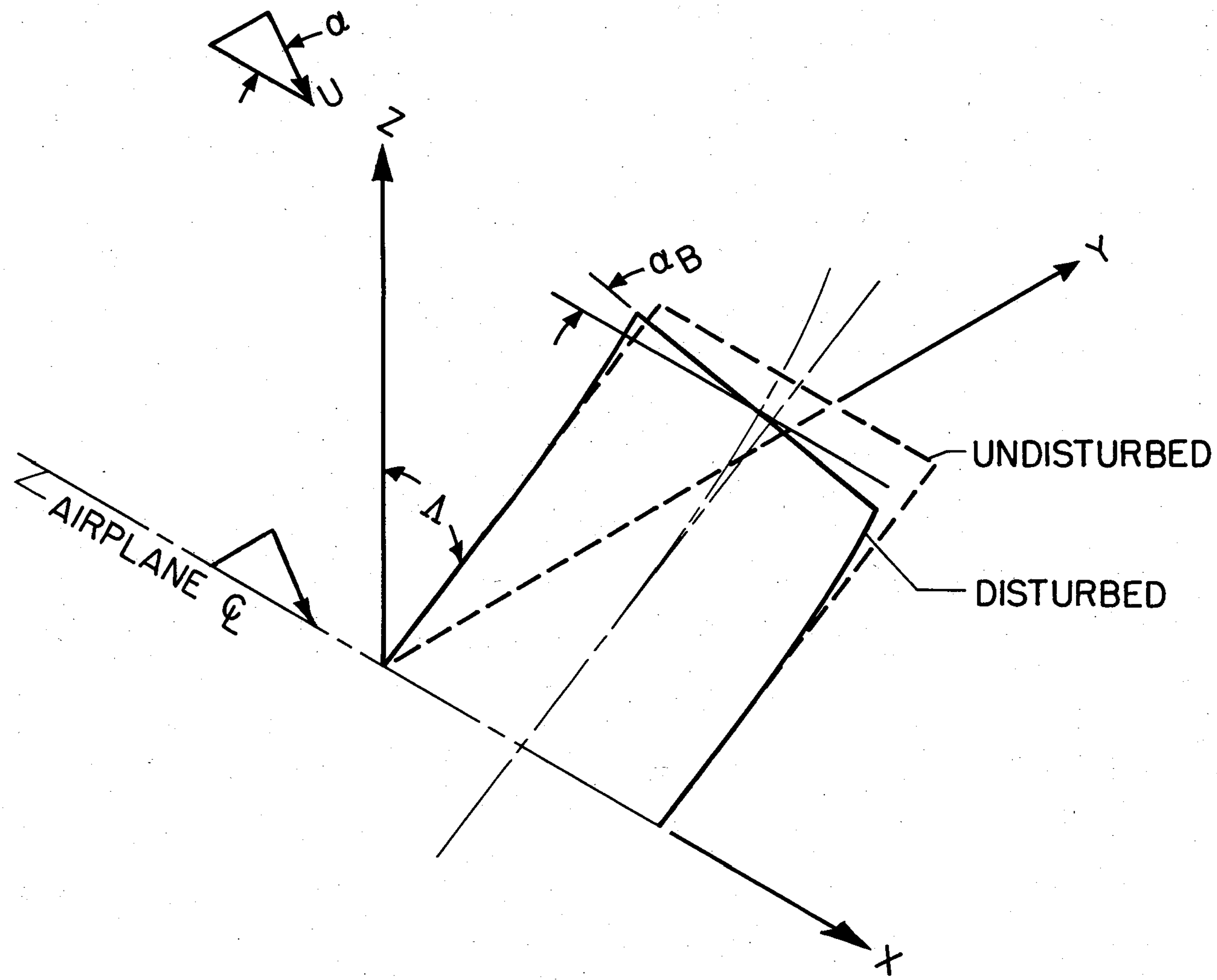
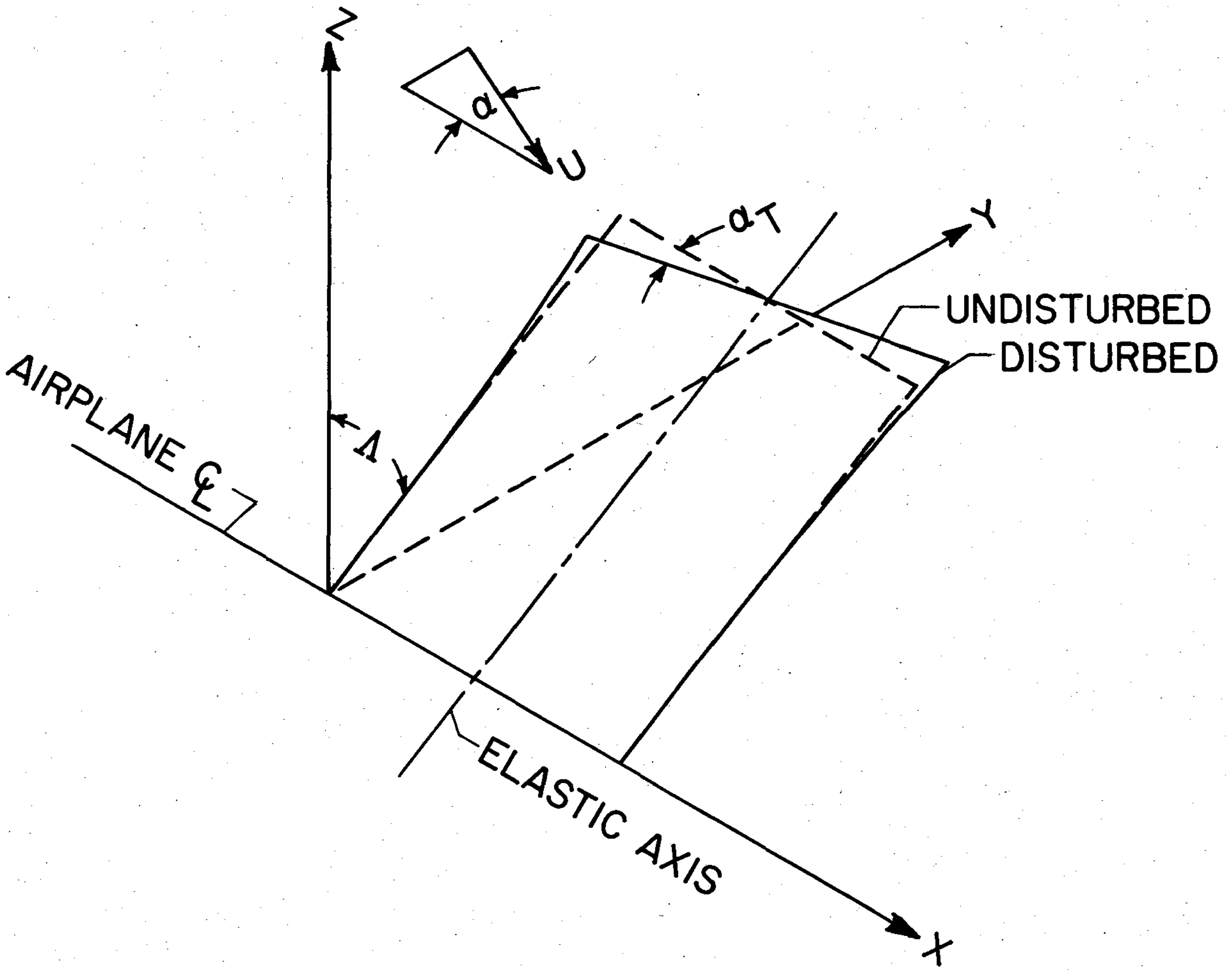
Figure 3.- Variation of aileron reversal speed with aileron span.



(a) Bending.

NASA

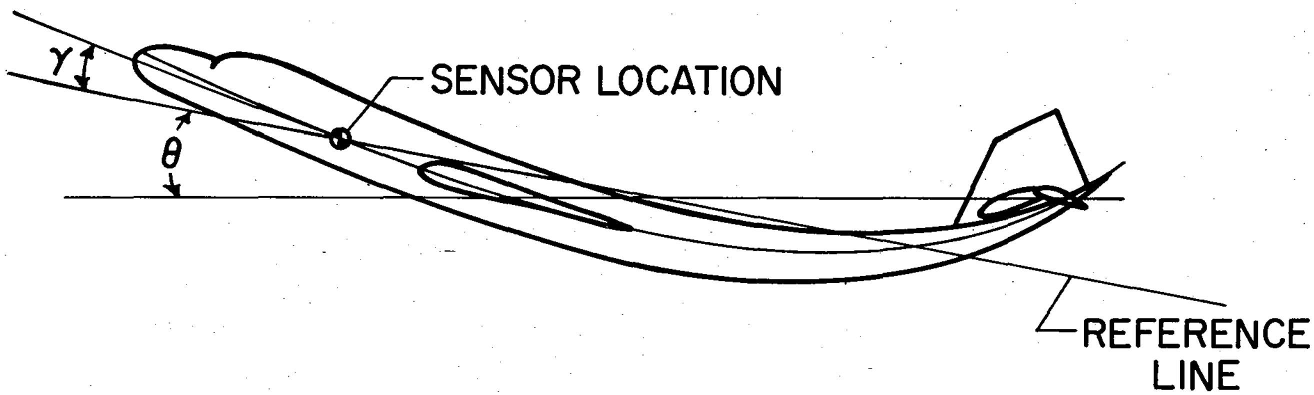
Figure 4.- Angle of attack change due to structural deformation of swept vertical fin.



(b) Twist.

Figure 4.- Concluded.

NASA

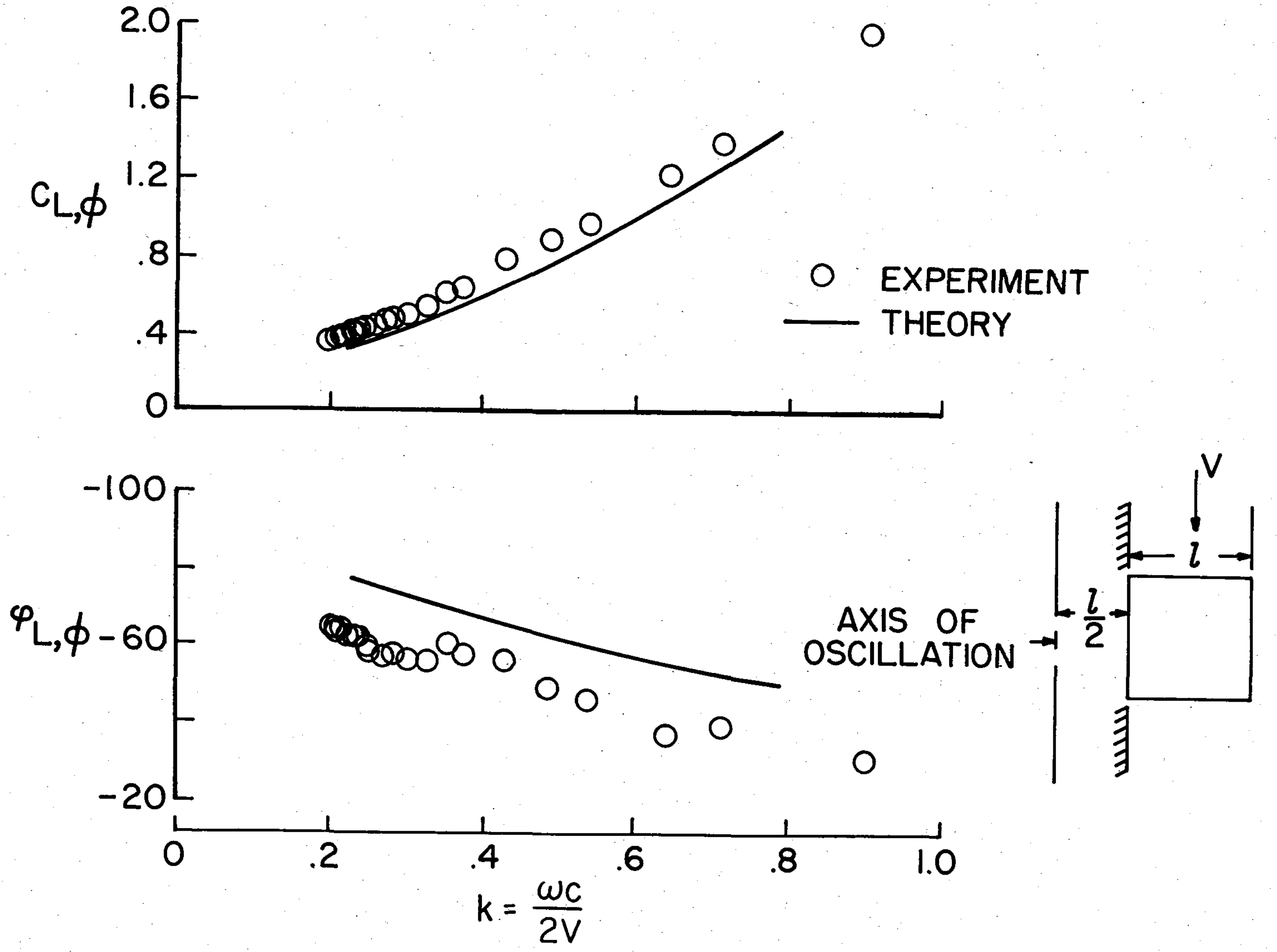


θ PITCH ANGLE

γ ADDITIONAL INCLINATION OF SENSOR DUE TO DEFORMATION

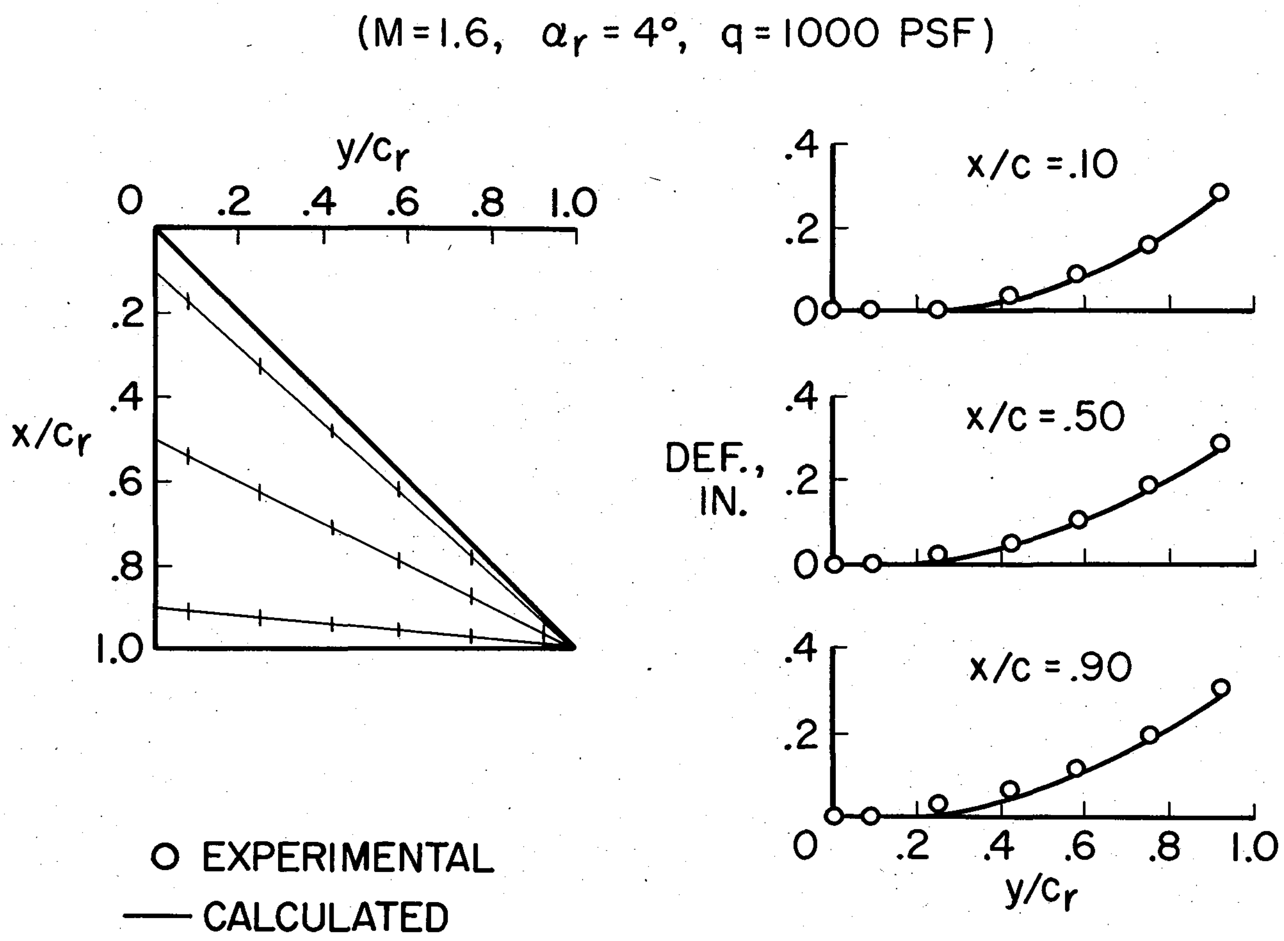
NASA

Figure 5.- Effect of structural deformation on automatic control sensor.



NASA

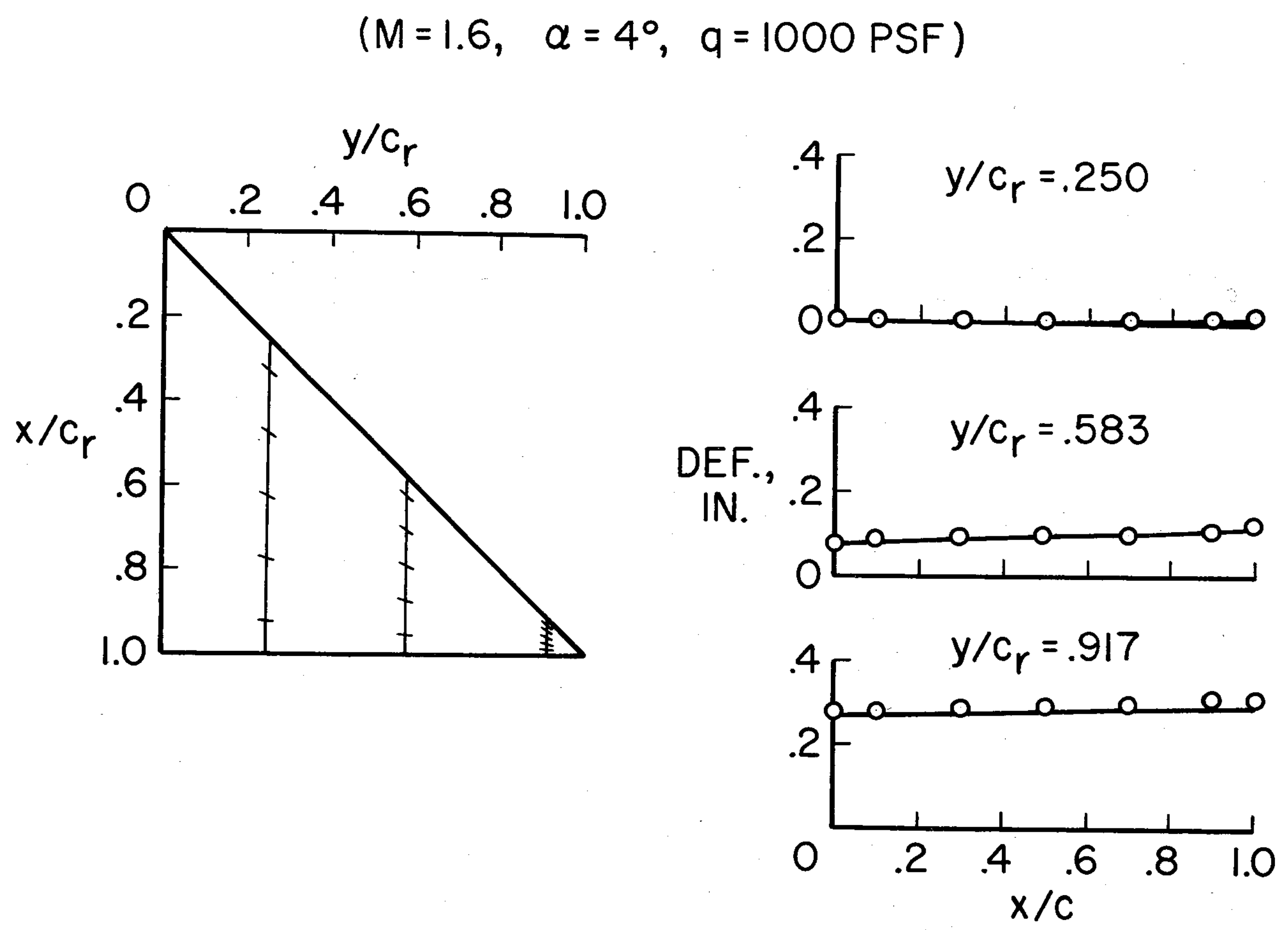
Figure 6.- Experimental and calculated lift on a wing oscillating in roll.



(a) Spanwise.

NASA

Figure 7.- Experimental and calculated deformation of a 45° delta wing.



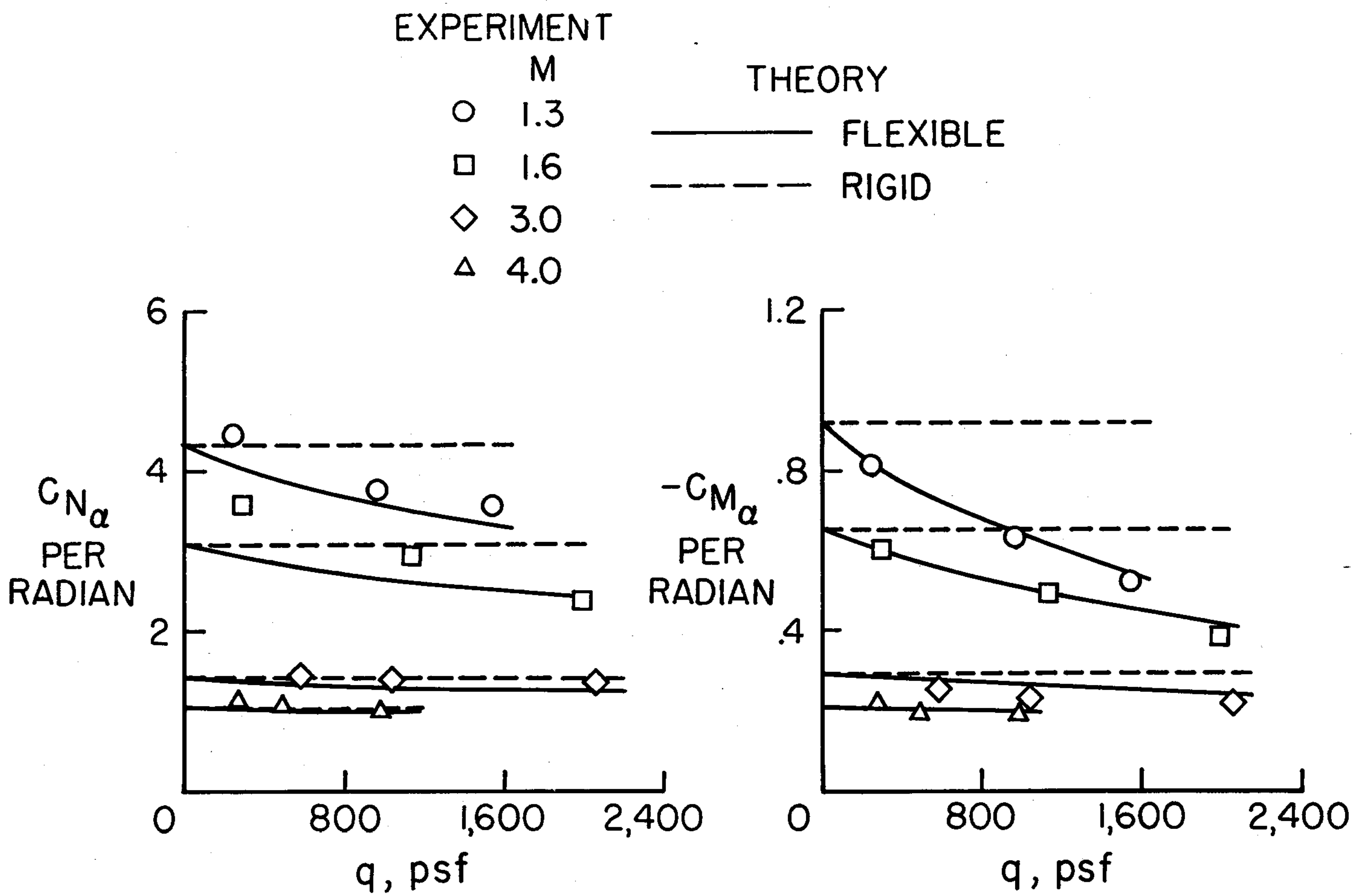
(b) Chordwise.

NASA

Figure 7.- Concluded.

L 1 4 4 6

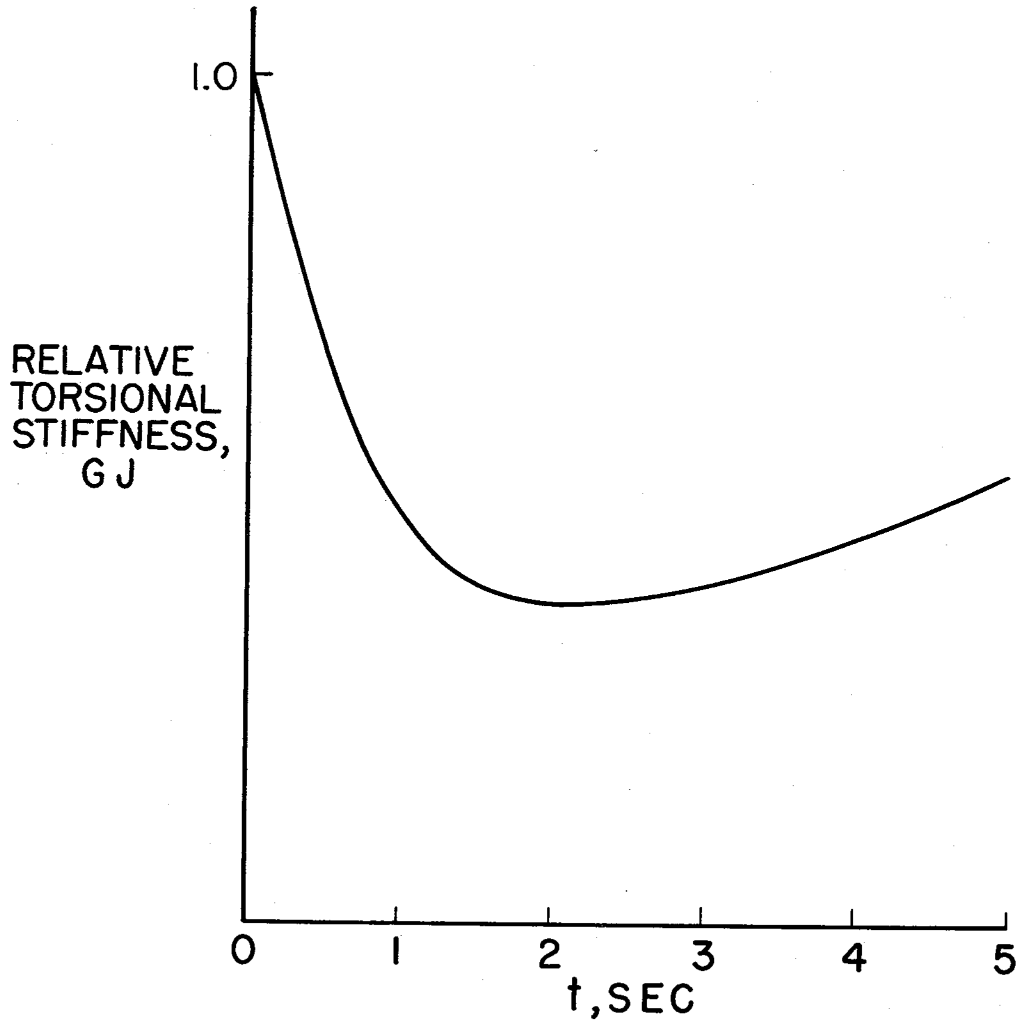
4M



NASA

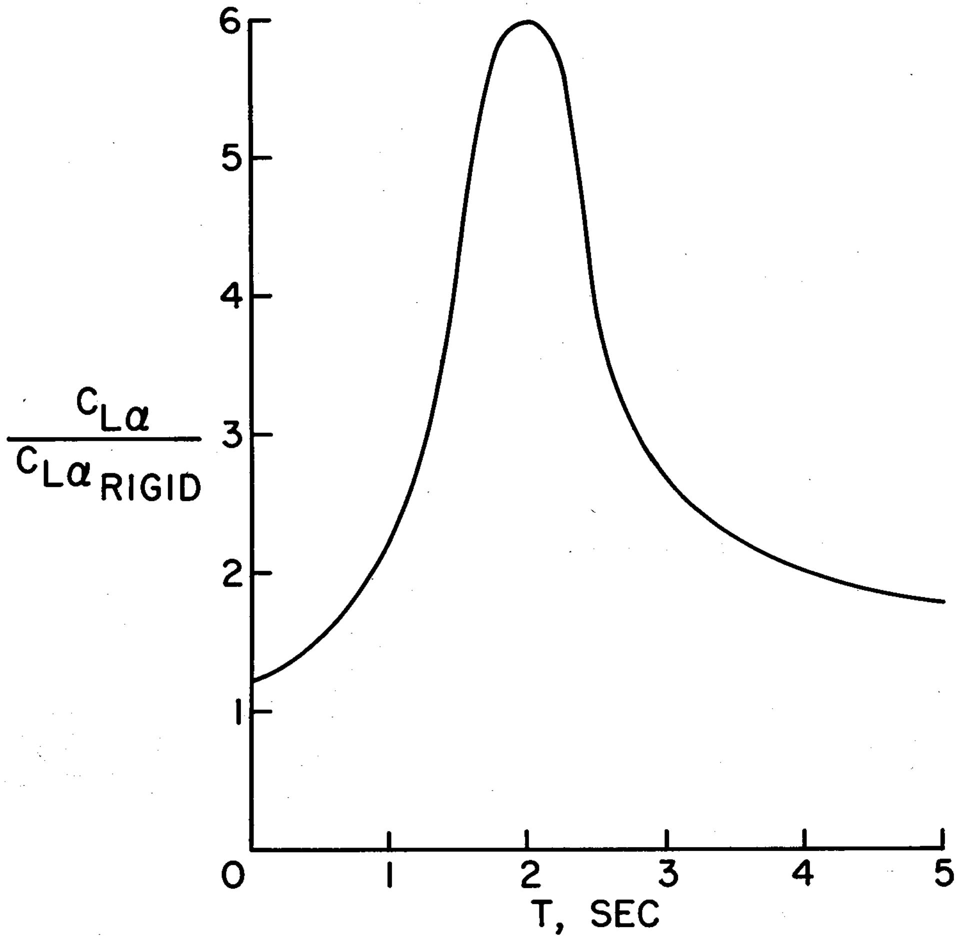
Figure 8.- Experimental and calculated normal force and pitching moment coefficients on a flexible 45° delta wing.

L 1 4 4 6



NASA

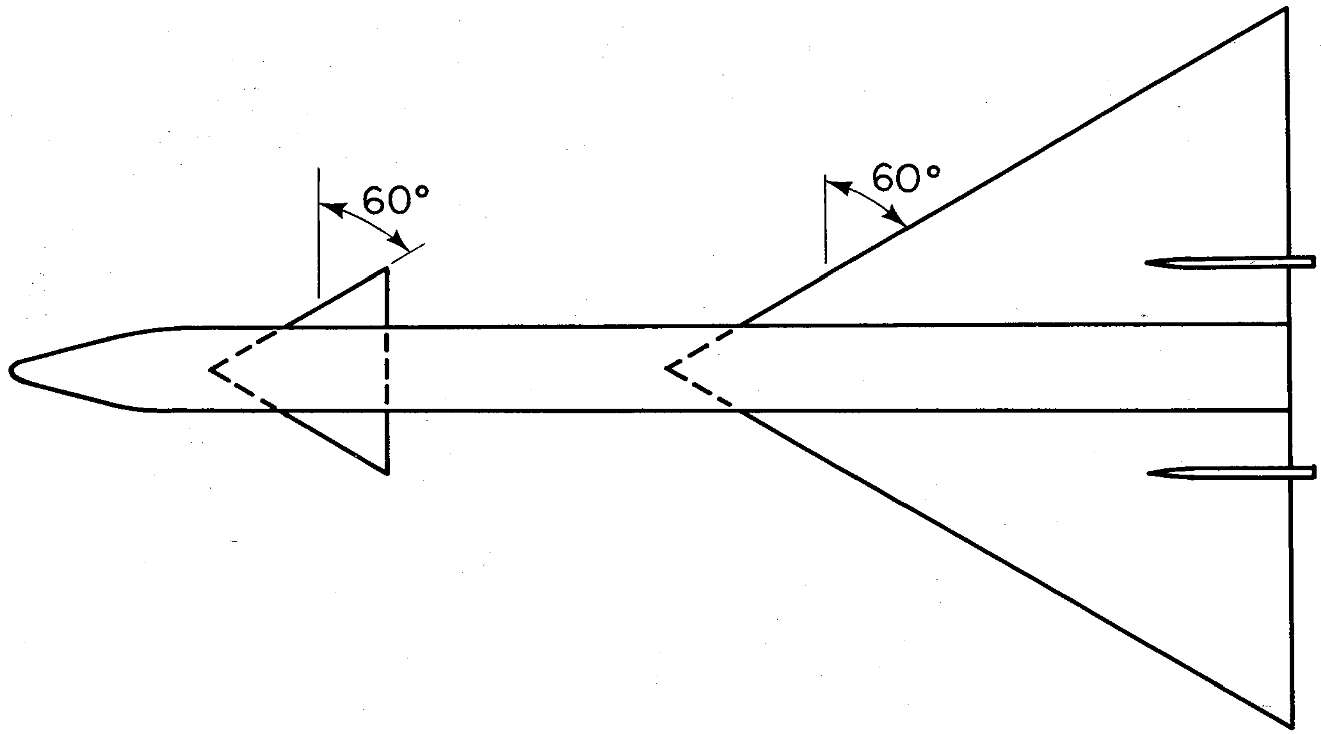
Figure 9.- Loss of torsional stiffness with time.



Transient thermal effects on torsion mode.

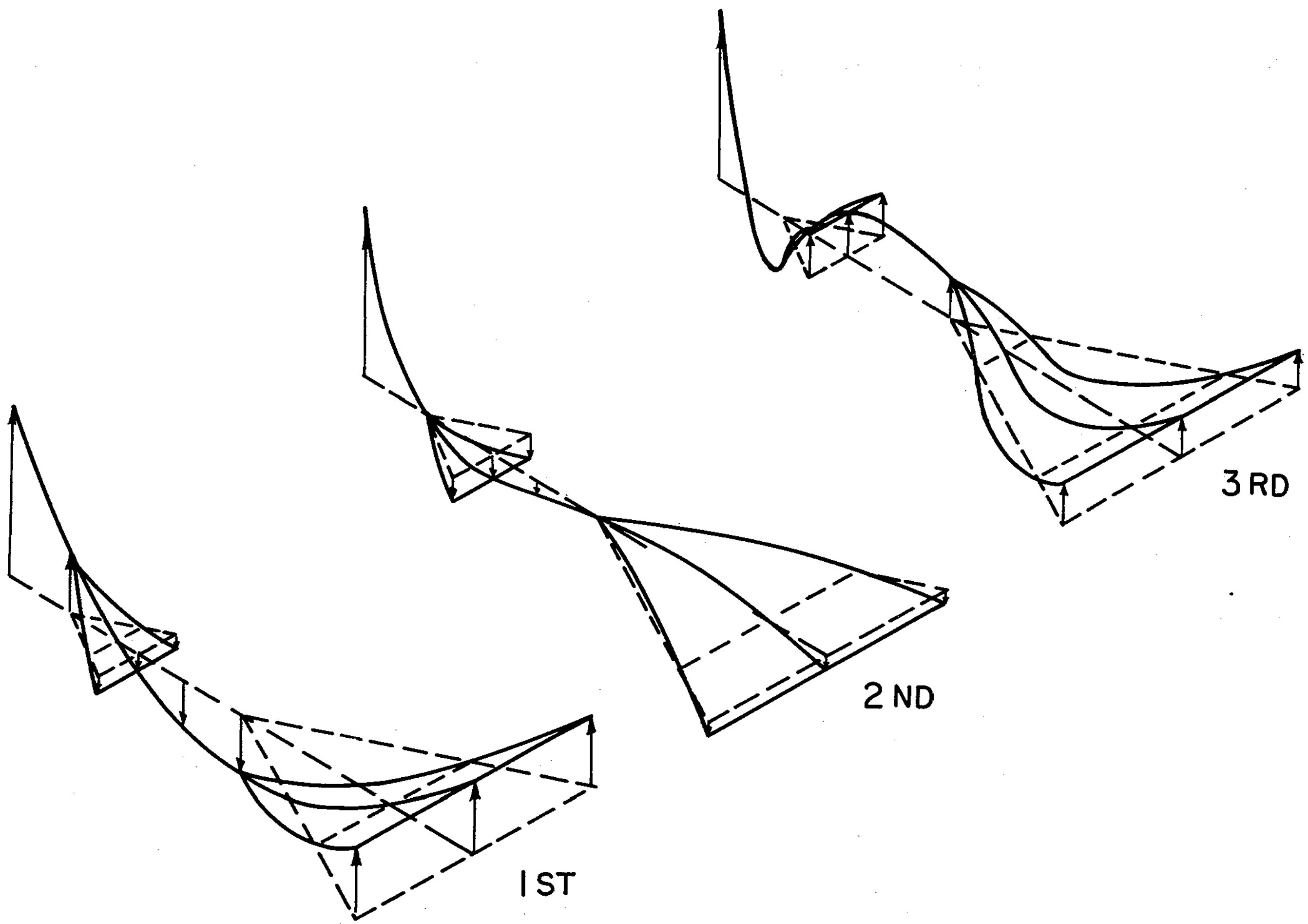
NASA

Figure 10.- Variation of lift curve slope with time.



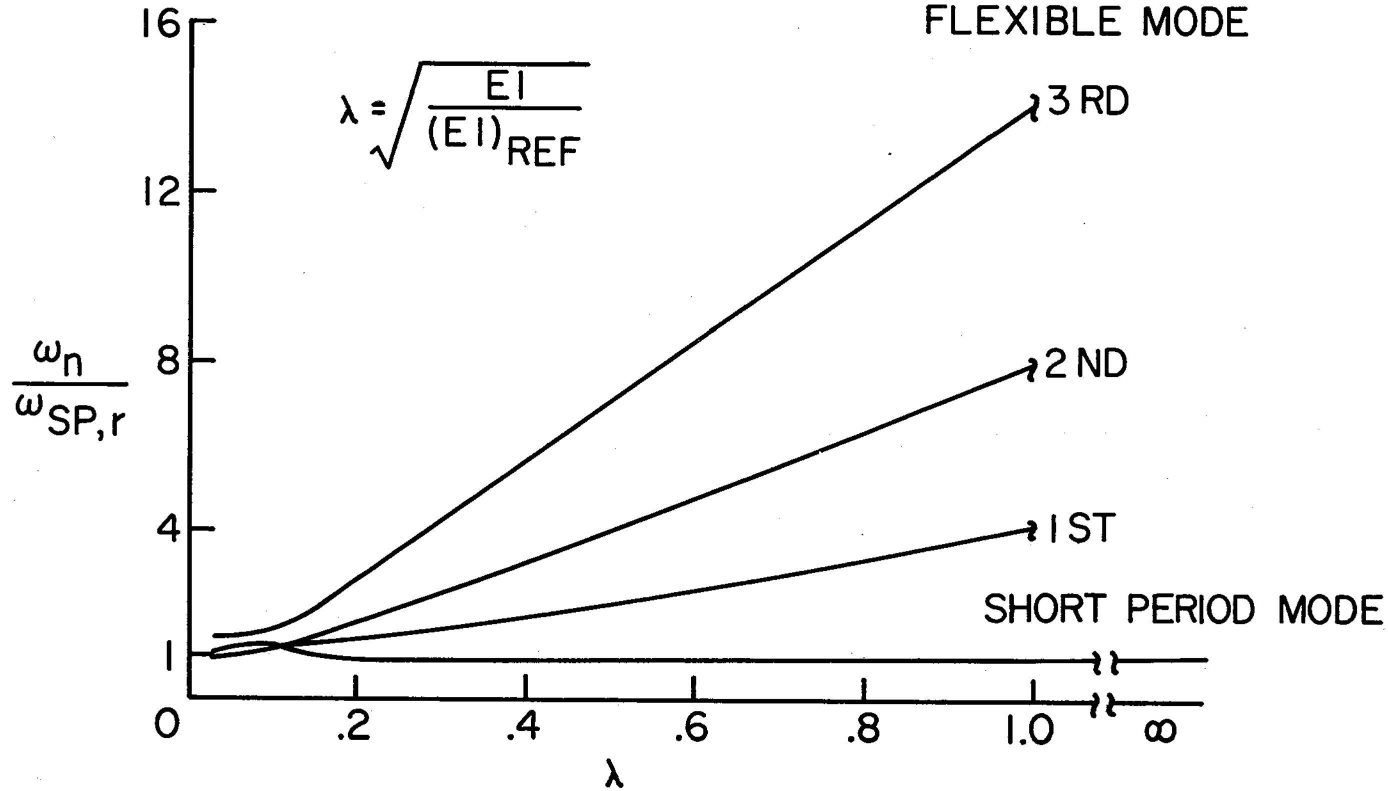
NASA

Figure 11.- Supersonic transport configuration.



NASA

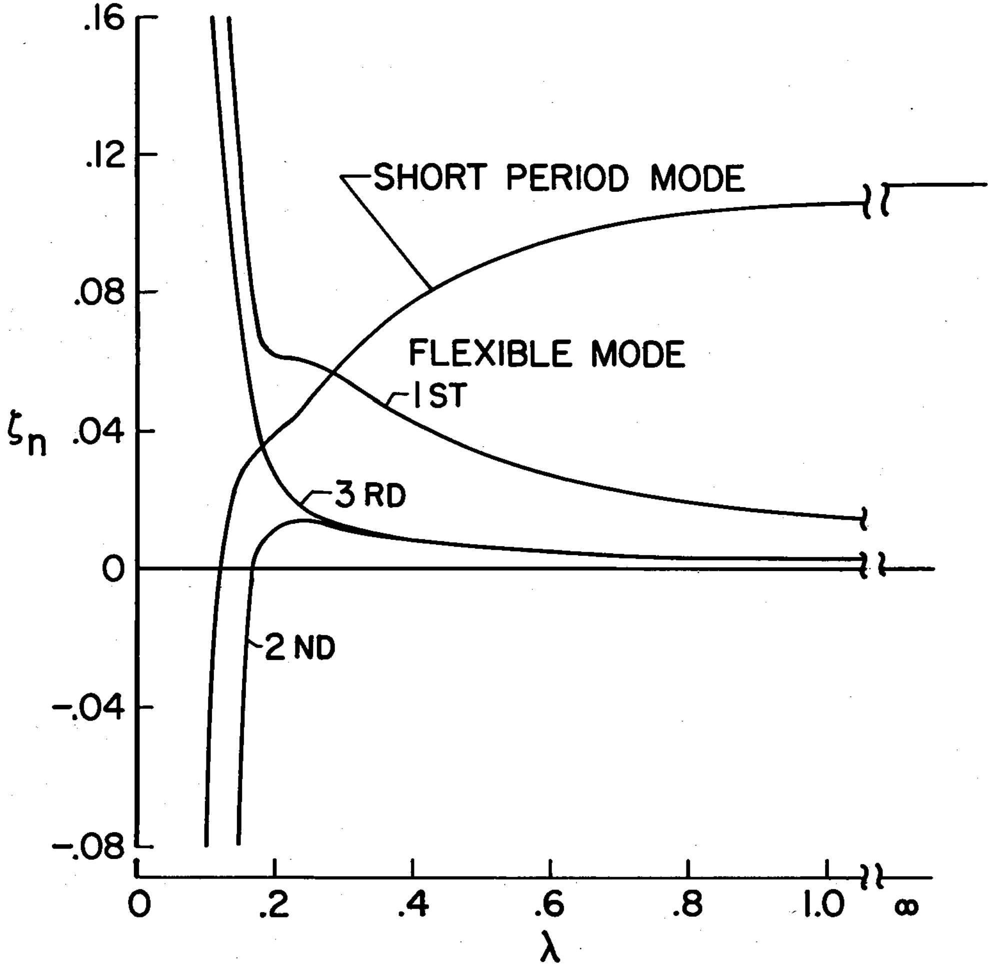
Figure 12.- Mode shapes used in supersonic transport analysis.



(a) Modal frequencies.

NASA

Figure 13.- Effect of fuselage stiffness on modal characteristics of supersonic transport.

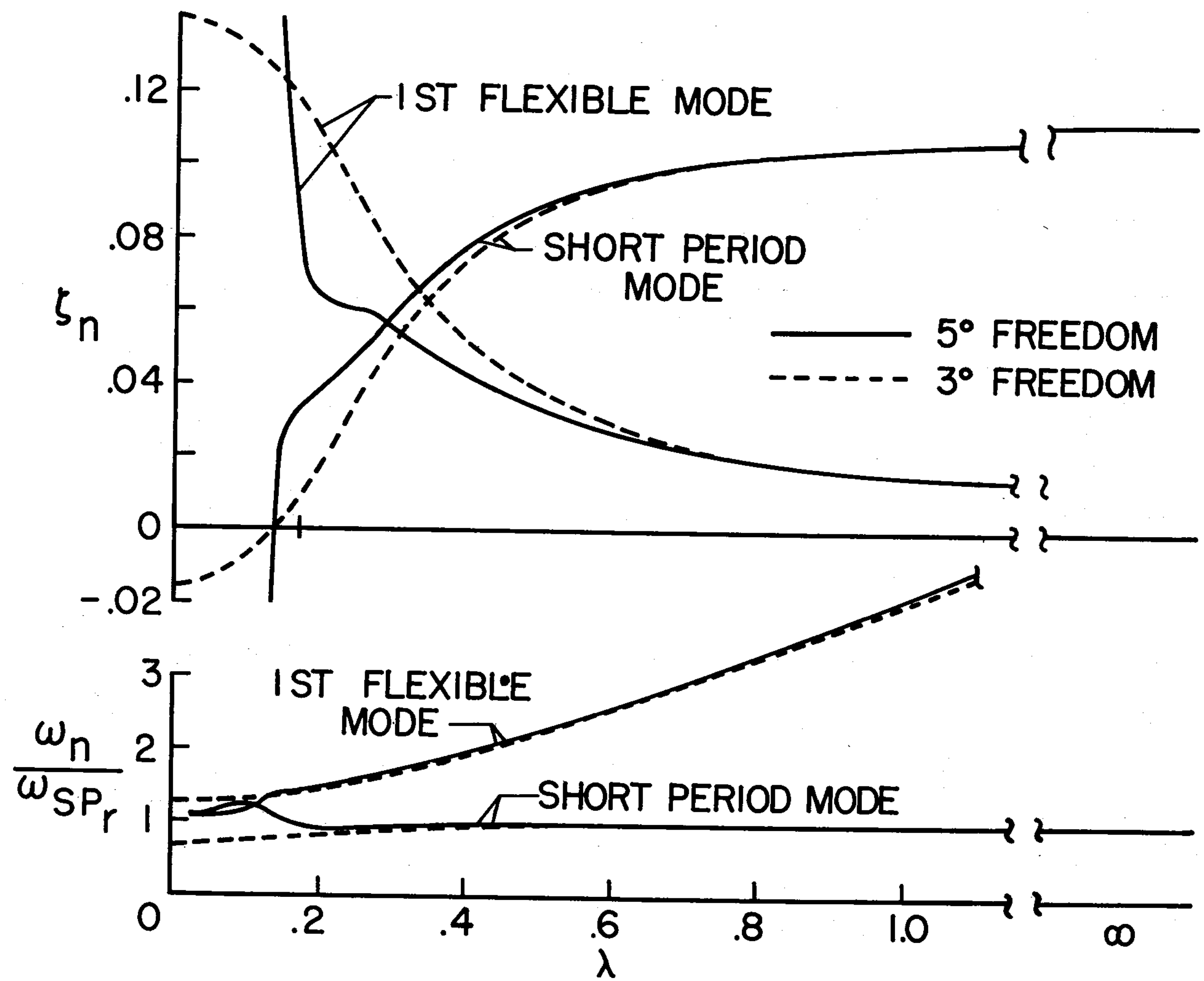


(b) Modal damping ratios.

Figure 13.- Concluded.

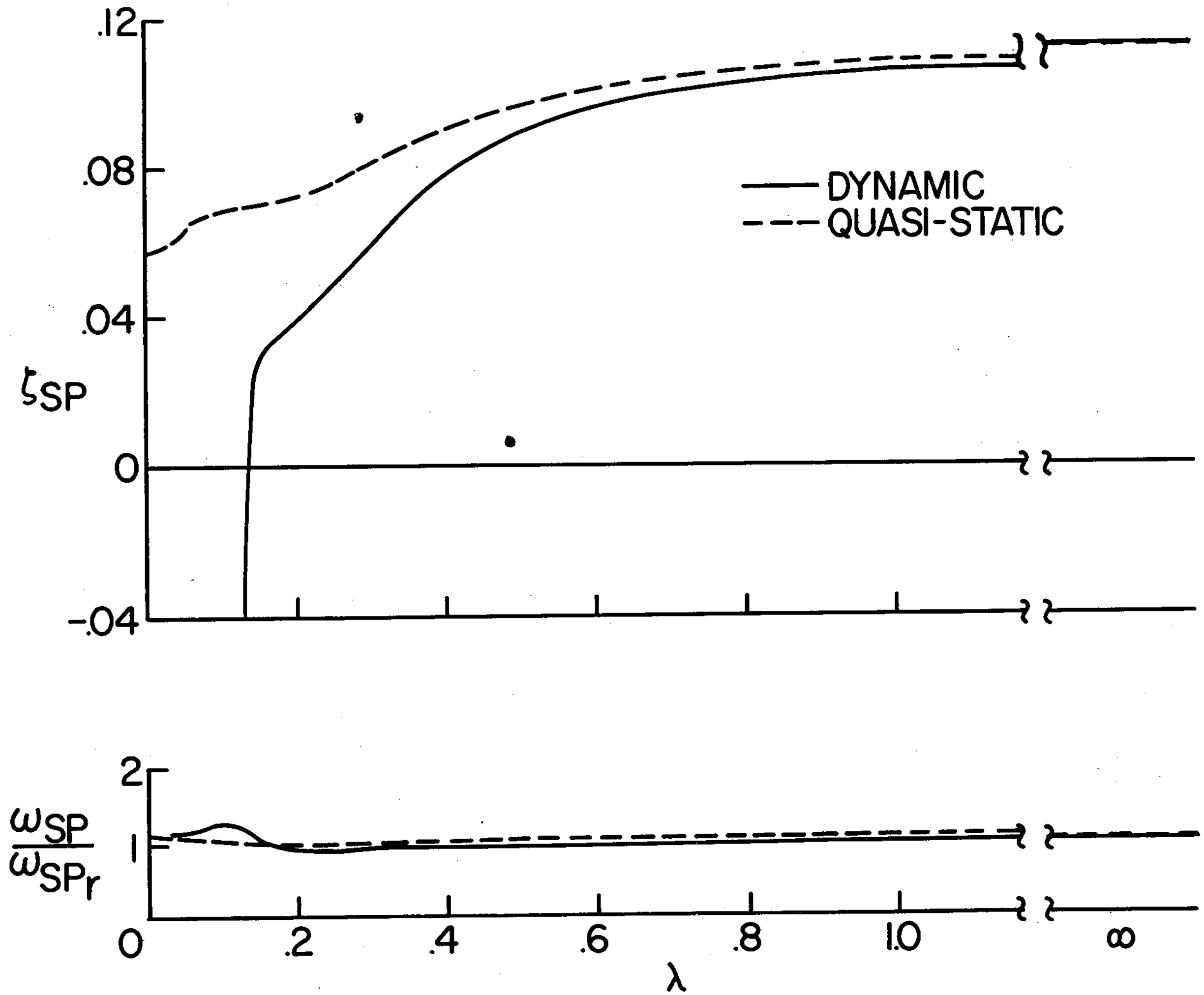
L1446

5M



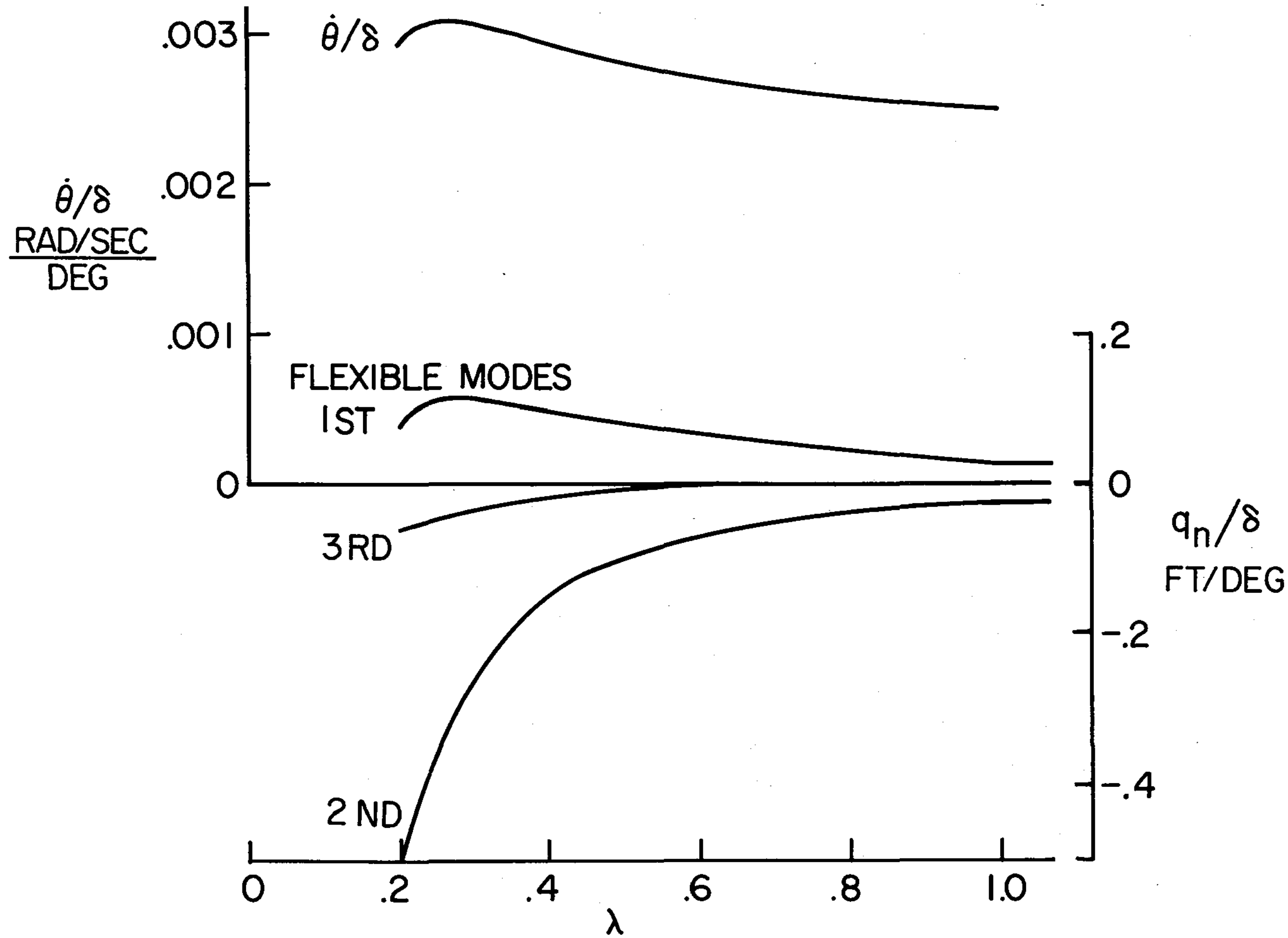
NASA

Figure 14.- Effect of reducing the number of modes on stability characteristics.



NASA

Figure 15.- Effect of ignoring dynamic flexibility on short period stability characteristics.



NASA

Figure 16.- Effect of fuselage stiffness on static control responses.