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# RESEARCH MEMORANDUM

for the

Bureau of Aeronautics, Navy Department

## CALCULATION OF WING BENDING MOMENTS AND TAIL LOADS RESULTING FROM THE JETTISON OF WING TIPS DURING A SYMMETRICAL PULL-UP

By

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NATIONAL ADVISORY COMMITTEE
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# CALCULATION OF WING BENDING MOMENTS AND TAIL LOADS RESULTING FROM THE JETTISON OF WING TIPS DURING A SYMMETRICAL PULL-UP

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## SUMMARY

A preliminary analytical investigation was made to determine the feasibility of the basic idea of controlled failure points as safety valves for the primary airplane structure. The present analysis considers the possibilities of the breakable wing tip which, in failing as a weak link, would relieve the bending moments on the wing structure.

The analysis was carried out by computing the time histories of the wing and stabilizer angle of attack in a 10g pull-up for an XF8F airplane with tips fixed and comparing the results with those for the same maneuver, that is, elevator motion but with tips jettisoned at 8g.

The calculations indicate that the increased stability accompanying the loss of the wing tips reduces the bending moment an additional amount above that which would be expected from the initial loss in lift and the inboard shift in load. The vortex shed when the tips are lost may induce a transient load requiring that the tail be made stronger than otherwise.

## INTRODUCTION

At the request of the Bureau of Aeronautics, Navy Department, an analytical investigation was undertaken to determine the feasibility of designing "safety valves" into the structure of an airplane. These safety valves were to be in the form of controlled failure points which would be incorporated into the structure in such a way that critical loads on the most vital parts of the primary structure would be relieved by the failure of a "weak link" located at a relatively less vital part of the structure.

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One device which has been considered is a breakable wing tip which would fail in a symmetrical pull-up at a predetermined acceleration. From a consideration of lift and lift distribution, it is apparent that the loss of wing tips would, at a given angle of attack, cause a decrease in wing bending moment due to both an inboard shift of the center of load and a loss in the value of lift because of a reduced aspect ratio and wing area. However, the effect of the loss of wing tips upon the angles of attack ultimately reached and upon the loads on the horizontal tail requires a study of the motions of the airplane.

In order to investigate some of the possibilities of the breakable wing tip as a safety valve, computations were made for a Grumman XF8F airplane. The purpose of this report is to present these results.

## METHOD AND RESULTS

The computations were carried out for two center-of-gravity locations at 24.6 and 30.6 percent of the original mean aerodynamic chord. It was assumed that a tip area equivalent to 10.7 percent of the original wing area was breakable; this corresponded to 17.6 percent of the wing span. The geometric and aerodynamic characteristics of the airplane with tips fixed and with tips off are presented in table I. The aerodynamic characteristics given in table I were obtained by using references 1 and 2 as guides and suitably modifying the results to apply specifically to the XF8F.

The effect of the loss of wing tips upon the maximum wing angle of attack reached and the magnitude of the horizontal tail loads was determined by computing the time histories of wing and tail angles of attack for a pull-up corresponding to an increment of 10g and comparing the results with the angles of attack which would be reached if it were assumed that the tips were jettisoned at a load-factor increment of 8. For the tips-fixed case, the time history of wing angle of attack was determined for a unit jump elevator deflection in which the elevator was instantaneously deflected $1^{\circ}$ and then held constant. By proportionality, the amount of elevator required to attain an increment in acceleration of 10g was determined. With the value of elevator deflection so determined and the type of motion described, the time histories of the increments in wing and tail angles of attack were computed both for the case of wing tips fixed and wing tips jettisoned at 8g. The time histories of the change in the wing angle of attack and the change in wind angle on the stabilizer are shown in figures 1 and 2 for both center-of-gravity positions. The details of the computations are presented in appendix I.

The change in wind angle at the stabilizer is used in figure 2 inasmuch as neither the initial trim angle of attack nor the effective angle of attack due to control deflection are pertinent to the comparison of maximum change in tail loads attained in each case. The results shown in figure 2, however, include the effect of a transient tail angle of attack associated with the sudden change in lift and lift distribution on the wing. The method of computation is included in appendix II.

Figure 3(a) shows the load distribution over the wing with tips on at a load factor of 8 and the load distribution at the same angle of attack with tips off. The amount of lift shed by the wing is indicated. Figure 3(b) shows, pictorially, the downstream movement of the trailing and shed vortices for different time intervals after being shed from the wing. The time history of the incremental angle of attack at the tail is shown in figure 3(c). The results shown in figure 3(c) were obtained by considering each span loading shown in figure 3(a) separately and replacing the actual distribution by lifting elements of different span lengths. The downwash was then determined by usual methods considering the bound, trailing, and free vortex. Further details of the computations for the transient are presented in appendix II.

Figure 4 shows the time histories of the wing bending moment in the form of an index so that a comparison with the original airplane bending moment for the tips-fixed pull-up, shown by the dashed line, could be readily made. The bending-moment index is the ratio of the instantaneous root bending moments of the tips-off wing to the tips-on wing assuming an initial elevator motion sufficient to give a load factor of 10 for the tips-on case.

## DISCUSSION

The time histories of the increment in wing angle of attack (fig. 1) indicate that the loss of wing tips causes an increase in stability with the result that the maximum angle of attack ultimately reached by the wing is lower than that which would have been reached with tips fixed. This provides an additional safety effect which is less important for the aft location of the center of gravity than the forward position. Figure 2 shows that the angle of attack of the tail is not appreciably changed from the value which would have been reached with tips fixed.

The effect of the breakable wing tip in preventing the attainment of critically high bending moments is evident from the results shown in figure 4. The reduction in bending moment after the tips break is due to a combination of an immediate inboard shift in the center of pressure and the loss of lift as well as the subsequent reduction in the angle of attack reached due to the increased stability. For the airplane investigated, the reduction in bending moment due to the change in stability is not as important as the other, although from figure 1(a) it is seen that, for the forward center-of-gravity position, a decrement of about 1g would result from the increase in stability alone. For the particular conditions of the computations, it is seen in figure 4 that the maximum bending moment for the tips-off case is that existing just before the tip break.

The magnitude of the loads and bending moments on the tail is proportional to the angles of attack shown in the time histories of figure 2 since neither the geometric nor the aerodynamic characteristics of the tail are altered during the maneuvers. Thus it is apparent from figure 2 that, with a conservative transient effect included, no load relief would be experienced by the tail. Roughly, the tail loads correspond in magnitude to those for the 10g pull-up, even though the maneuver from the wing-stress standpoint is limited to an 8g pull-up.

The results presented for the case of constant elevator angle might be modified if the pilot were to move the elevator following the loss of the tips. Because of the infinite number of possibilities of elevator movement, this effect will be discussed in a qualitative manner only. Pushing forward on the stick immediately after the loss of the tips would reduce the wing load, but there would be the possibility of increasing the peak tail loads still further if the movement were made suddenly.

Pulling back on the stick following failure would increase the wing load throughout the maneuver. It would initially serve to reduce the peak tail load that follows the loss of the tips, but a general increase in the level of the tail loads would occur over the latter part of the maneuver. Whether or not this general increase would be above that for the tips-on case would depend upon the load factor associated with the additional pull on the stick. Since the same design load factor could be carried by the wings and since the bending moments on the wings would be less, the airplane might conceivably be considered stronger following the loss of the wing tips, provided the tail initially were designed with extra strength.

## CONCLUDING REMARKS

With regard to the use of breakable wing tips as a safety valve for the primary wing structure, the following statements may be made:

1. The reduction in bending moment on the wing following the loss of the tips arises mainly from the inboard shift of the center of pressure and to the loss in lift due to reduced area and aspect ratio. The increased stability which accompanies the loss of tips also results in a reduction in both the bending moment and load factor reached with a given elevator deflection. This reduction is larger for the forward center-of-gravity position than for the aft center-of-gravity position.

2. Following the loss of the tips, the wing would possibly have a greater load capacity insofar as bending is concerned.

3. The tail of an airplane incorporating breakable tips should be made stronger to withstand possible transient load effects resulting from the vortex shed with the loss of the wing tips.

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# APPENDIX I

## BASIS FOR COMPUTATION OF WING AND TAIL ANGLE OF ATTACK

As shown in reference 3, the computation of the wing and tail angle of attack for an infinitely fast control deflection is straightforward as the equation of motion is in the form of the familiar second-order linear differential equation

$$\ddot{\alpha} + K_1\dot{\alpha} + K_2\alpha = K_3\Delta\delta$$

in which $\alpha$ is the incremental angle of attack of the wing and the constants are determined by the aerodynamic and geometric characteristics of the airplane. Also in reference 3 a graphical method of performing a Du Hamel integration is presented by means of which the time history for a step-type of elevator motion may be converted into the solution for an arbitrary type of control motion.

Inasmuch as the time history of the wing angle of attack and rate of change of wing angle of attack for a finite but fast rate of control deflection are not appreciably different from those for an infinitely fast rate of control deflection, the step-type control deflection was assumed to simplify the computations. The solution of the equation for the wing angle of attack is

$$\alpha = e^{-\frac{K_1}{2}t}\left(C_1 \cos pt + C_2 \sin pt\right) + \frac{K_3}{K_2}\Delta\delta$$

and for the rate of change of wing angle of attack:

$$\dot{\alpha} = e^{-\frac{K_1}{2}t}\left(D_1 \cos pt + D_2 \sin pt\right)$$

in which

$$p = \sqrt{K_2 - \frac{K_1^2}{4}}$$

$$C_1 = \alpha(0) - \frac{K_3}{K_2}\Delta\delta$$

$$C_2 = \frac{\dot{\alpha}(0) + C_1\frac{K_1}{2}}{p}$$

$$D_1 = C_2 p - \frac{K_1}{2}C_1$$

$$D_2 = pC_1 - \frac{K_1}{2}C_2$$

and the values of the constants (reference 3) are:

$$K_1 = \frac{\rho V}{2m}\left[\left(\frac{dC_L}{d\alpha}\right)_t \frac{S_t x_t^2}{K_Y^2} n_t \left(\frac{K}{\sqrt{n_t}} + \frac{d\epsilon}{d\alpha}\right) + \frac{dC_L}{d\alpha} S\right]$$

$$K_2 = -\frac{\rho V^2}{2m}\left\{\frac{dC_m}{d\alpha}\frac{S^2}{K_Y^2 b} + \left(\frac{dC_L}{d\alpha}\right)_t n_t \frac{S_t x_t}{K_Y^2}\left[\left(1 - \frac{d\epsilon}{d\alpha}\right) - \frac{dC_L}{d\alpha}\frac{K}{\sqrt{n_t}}\frac{\rho}{2}\frac{S x_t}{m}\right]\right\}$$

$$K_3 = \frac{\rho V^2}{2m}\left[\frac{dC_{L_t}}{d\delta} n_t \frac{S_t x_t}{K_Y^2} - \frac{dC_{m_t}}{d\delta} n_t \frac{S_t^2}{b_t K_Y^2} - \left(\frac{dC_L}{d\alpha}\right)_t \frac{dC_{L_t}}{d\delta}\frac{K n_t^2}{\sqrt{n_t}}\frac{\rho}{2}\frac{x_t^2 S_t^2}{m K_Y^2}\right]$$

For the tail angle of attack the solution is

$$\Delta\alpha_t = \left[\Delta\alpha\left(1 - \frac{d\epsilon}{d\alpha} - \frac{dC_L}{d\alpha}\frac{\rho}{2}\frac{S}{m}\frac{x_t}{\sqrt{n_t}}\right) - \dot{\alpha}\frac{x_t}{V}\left(\frac{d\epsilon}{d\alpha} + \frac{1}{\sqrt{n_t}}\right) + \frac{d\alpha_t}{d\delta}\Delta\delta\right]$$

The wing and tail loads are obtained by use of the lift equation for the wing

$$\Delta L = \Delta\alpha\left(\frac{dC_L}{d\alpha}\right)_w qS$$

and for the tail

$$\Delta L_t = \Delta\alpha_t\left(\frac{dC_L}{d\alpha}\right)_t n_t q S_t$$

In the computations it was assumed that the airplane was in steady trim flight initially so that $\alpha(0)$ and $\dot{\alpha}(0)$ in the definition of constants $C_1$ and $C_2$ were zero.

For the maneuvers in which the tips were jettisoned at 8g the constants $K_1$ and $K_2$ were computed by using the aerodynamic and geometric characteristics of the airplane less wing tips. The values of $\alpha(0)$ and $\dot{\alpha}(0)$ in the constants $C_1$ and $C_2$, respectively, have the values which exist the instant before the wing tips are jettisoned. The elevator angle was assumed to be held at a constant value throughout the maneuver.

For the sake of consistency, the aerodynamic characteristics which changed for wing tips on and off $\left(dC_L/d\alpha,\ dC_m/d\alpha,\ \text{and}\ d\epsilon/d\alpha\right)$ were determined by means of reference 2 for both the wing tips fixed and wing tips off. (The values computed using reference 2 for the wings-fixed configuration compared favorably with those measured in the wind-tunnel tests of reference 1.)

# APPENDIX II

## CALCULATION OF TRANSIENT

The incremental angle of attack at the tail due to the abrupt loss in wing lift was calculated by approximating the loss in lift by step elements and evaluating the effect of the bound, shed, and trailing vortices for various positions of the vortex in its travel downstream from the wing. For the approximate results desired on this question, no attempt was made to consider the lag in circulation and therefore the results may be more conservative than otherwise.

The following equations were used to evaluate the downwash velocity $w$ on the tail at the line of symmetry.

For the bound vortex of each lifting element,

$$\frac{w}{V} = \frac{1}{2\pi x_o}\left(\Delta C_L \frac{c}{b}\right) \cos \alpha_o$$

in which $x_o$ (see fig. 5) is the distance from the wing quarter chord to the tail quarter chord in semispans, $\alpha_o$ is that angle formed by the wing quarter-chord line and a line from the end of the bound vortex to the tail quarter-chord point at the line of symmetry, and $\Delta C_L \frac{c}{b}$ is the strength of each lifting element used to approximate the span load distribution.

For the shed vortex of each lifting element

$$\frac{w}{V} = \frac{1}{2\pi x_n}\left(\Delta C_L \frac{c}{b}\right) \cos \alpha_n$$

in which $x_n$ is the distance from the shed vortex line to the tail, as shown in figure 5, and $\alpha_n$ is the angle between the shed vortex line and a line to the center of the tail quarter-chord line from the end of the shed line.

For the trailing vortices of each lifting element

$$\frac{w}{V} = \frac{\Delta C_L \frac{c}{b}}{2\pi y} (\cos \beta_o + \cos \gamma_n)$$

in which $y$ is the span of the lifting element in wing semispans; $\beta_o$ is the angle formed between the trailing vortex and a line from the origin of the trailing vortex to the tail; similarly, $\gamma_n$ is the angle formed by the trailing vortex line and a line from the end of the trailing vortex to the tail quarter-chord point.

## REFERENCES

1. Riebe, John M., and Spear, Margaret F.: Wind-Tunnel Tests of a 1/5-Scale Powered Model of the Grumman XF8F-1 Airplane. II - Longitudinal Stability and Control - TED No. NACA 2344. NACA MR No. L5E16, Bur. Aero., 1945.

2. Gilruth, R. R., and White, M. D.: Analysis and Prediction of Longitudinal Stability of Airplanes. NACA Rep. No. 711, 1941.

3. Pearson, Henry A.: Derivation of Charts for Determining the Horizontal Tail Load Variation with Any Elevator Motion. NACA Rep. No. 759, 1943.

# TABLE I

## AIRPLANE CHARACTERISTICS AND SYMBOLS

| Symbol | Definition | Value original airplane | Value airplane tips off |
|---|---|---|---|
| $S$ | Wing area, sq ft | 244 | 217.9 |
| $W$ | Weight, lb | 8800 | 8800 |
| $K_Y$ | Radius of gyration, ft | 5.46 | 5.46 |
| $b$ | Wing span, ft | 35.5 | 29.2 |
| $\bar{c}$ | Mean chord, ft | 6.88 | 7.47 |
| $m$ | Airplane mass, $W/g$ | 273 | 273 |
| A.R. | Aspect ratio, $b^2/S$ | 5.16 | 3.92 |
| $\lambda$ | Taper ratio, root chord/tip chord | 2.27 | 1.888 |
| $S_t$ | Horizontal tail area, $ft^2$ | 52.2 | 52.2 |
| $b_t$ | Horizontal tail span, ft | 15.75 | 15.75 |
| $x_t$ | Tail length, from c.g. to a.c. of tail, ft (negative for conventional airplanes) | [a]-17.10<br>[b]-16.74 | [a]-17.10<br>[b]-16.74 |
| $\rho$ | Mass density of air, slugs/cu ft | 0.0020 | 0.0020 |
| $V$ | True airspeed, ft/sec | 500 | 500 |
| $K$ | Empirical constant denoting ratio of damping moment of complete airplane to damping moment of tail alone | 1.1 | 1.1 |
| $n_t$ | Tail-efficiency factor, $q/q_t$ | 1.0 | 1.0 |
| $(dC_L/d\alpha)_t$ | Rate of change of tail lift coefficient with angle of attack for isolated tail, per radian | 2.72 | 2.72 |

[a]c.g. at 24.6

[b]c.g. at 30.6

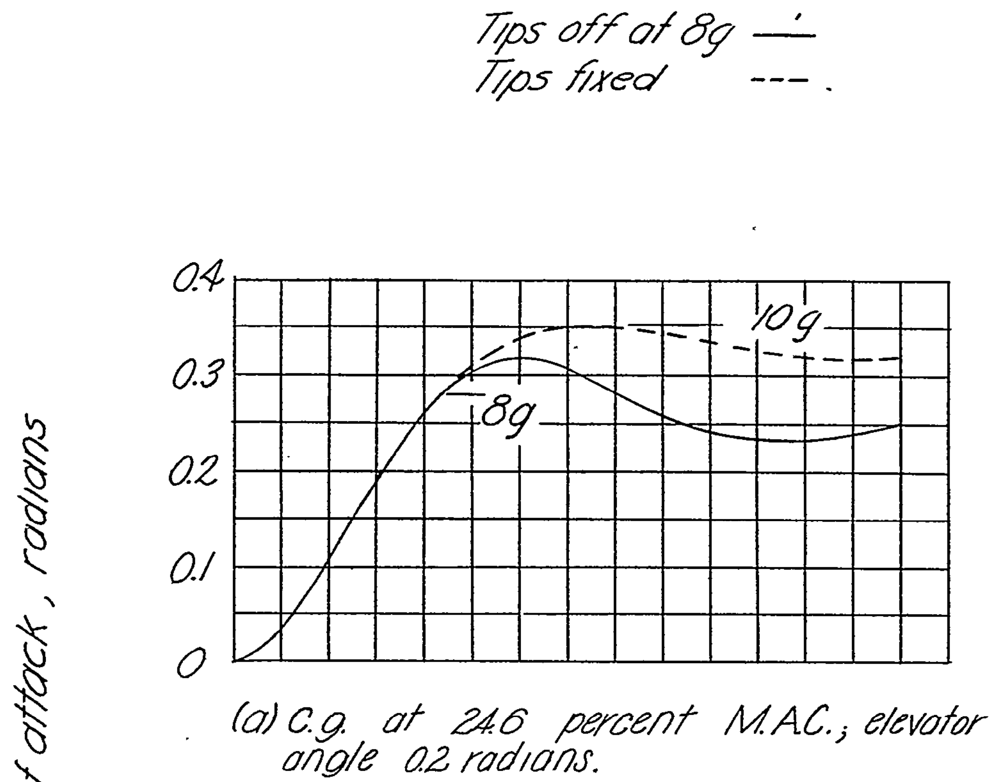
TABLE I - Concluded

AIRPLANE CHARACTERISTICS AND SYMBOLS - Concluded

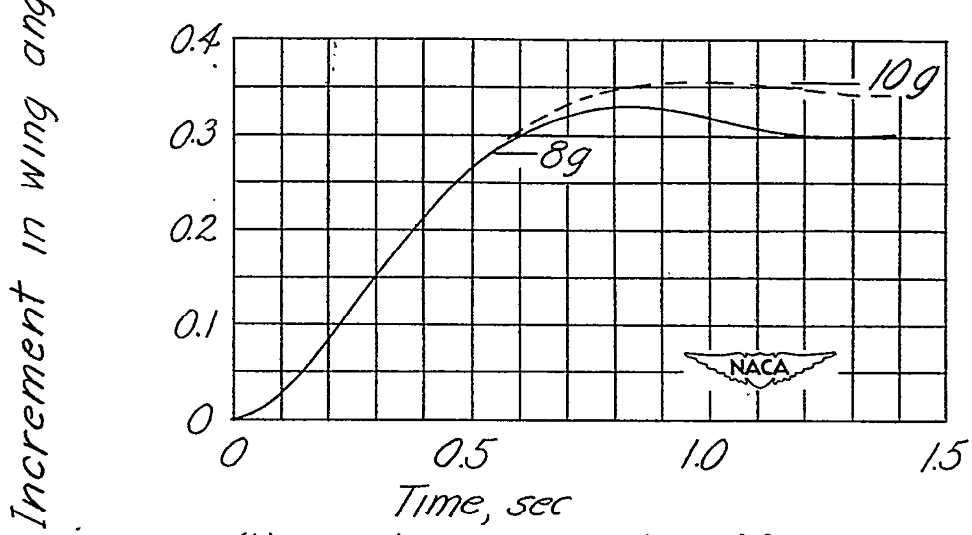
| Symbol | Definition | Value original airplane | Value airplane tips off |
|---|---|---|---|
| $dC_m/d\alpha$ | Rate of change of pitching-moment coefficient with angle of attack for airplane less tail, per radian | [a]0.288<br>[b]0.493 | [a]0.297<br>[b]0.468 |
| $d\epsilon/d\alpha$ | Rate of change of downwash angle at tail with angle of attack | 0.505 | 0.425 |
| $(dC_L/d\alpha)_w$ | Rate of change of wing lift coefficient with angle of attack, per radian | 4.09 | 3.47 |
| $dC_L/d\alpha$ | Rate of change of airplane lift coefficient with angle of attack, per radian | 4.38 | 3.85 |
| c | Local wing chord | | |
| q | Dynamic pressure | | |

[a]c.g. at 24.6

[b]c.g. at 30.6




(a) C.g. at 24.6 percent M.A.C.; elevator angle 0.2 radians.




(b) C.g. at 30.6 percent M.A.C.; elevator angle 0.145 radians.

Figure 1. - Time histories of wing angle of attack in abrupt pull-ups for wing tips fixed and tips jettisoned at 8g.

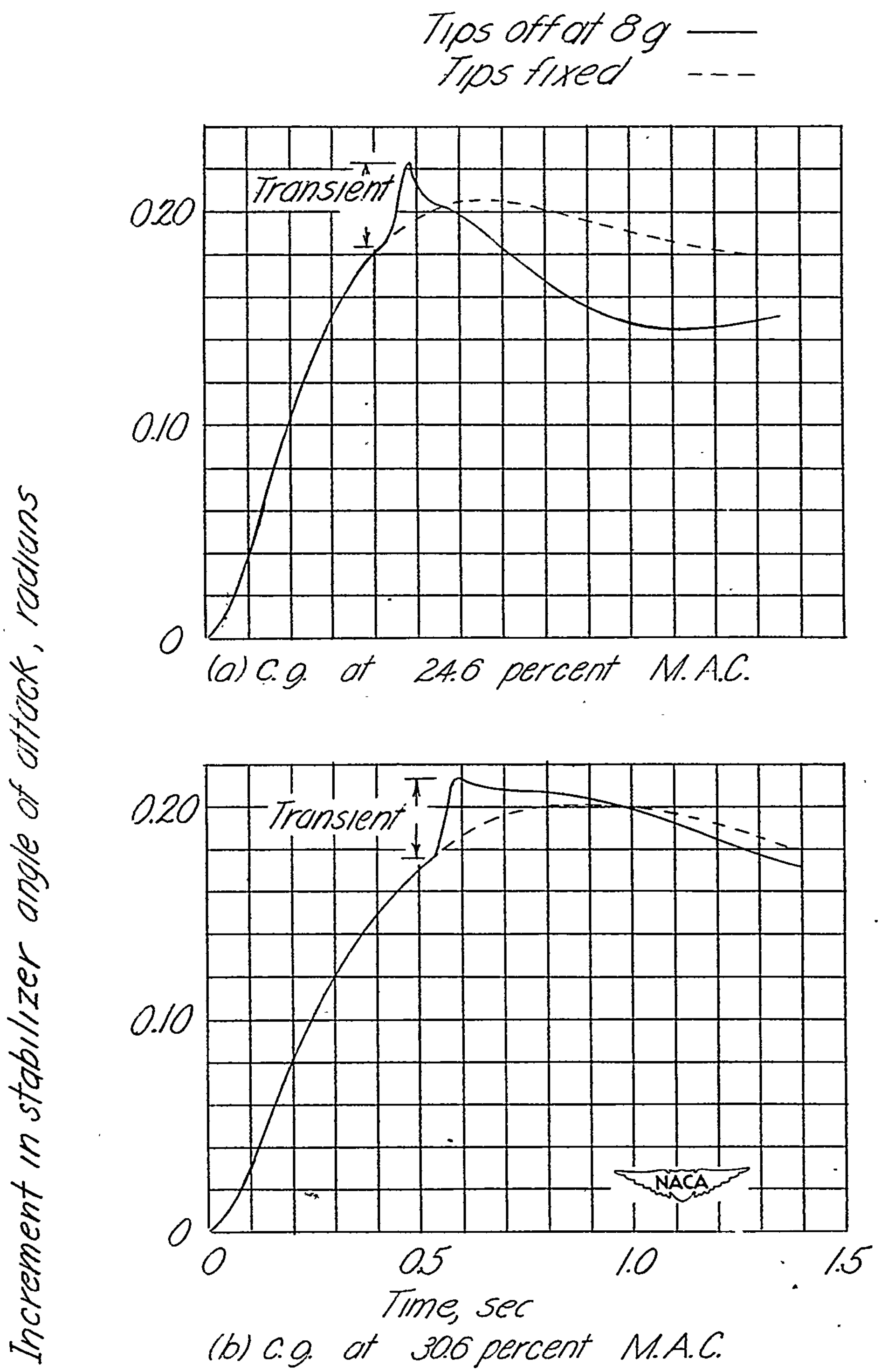



Figure 2.- Time histories of stabilizer angle of attack in abrupt pull-ups for wing tips fixed and tips jettisoned; at 8g.

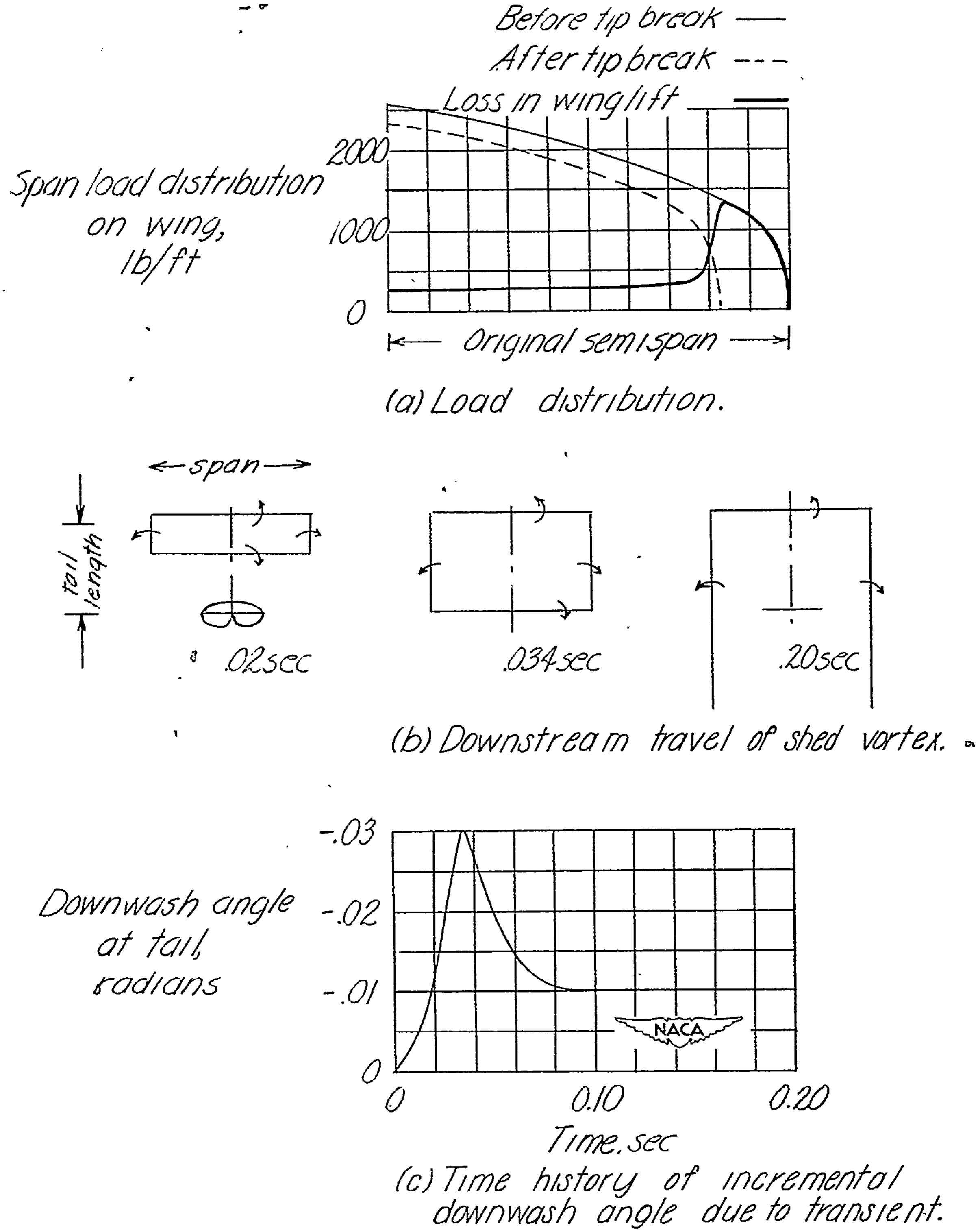



Figure 3.- Effect of sudden loss of wing tips on load distribution and tail downwash angle.

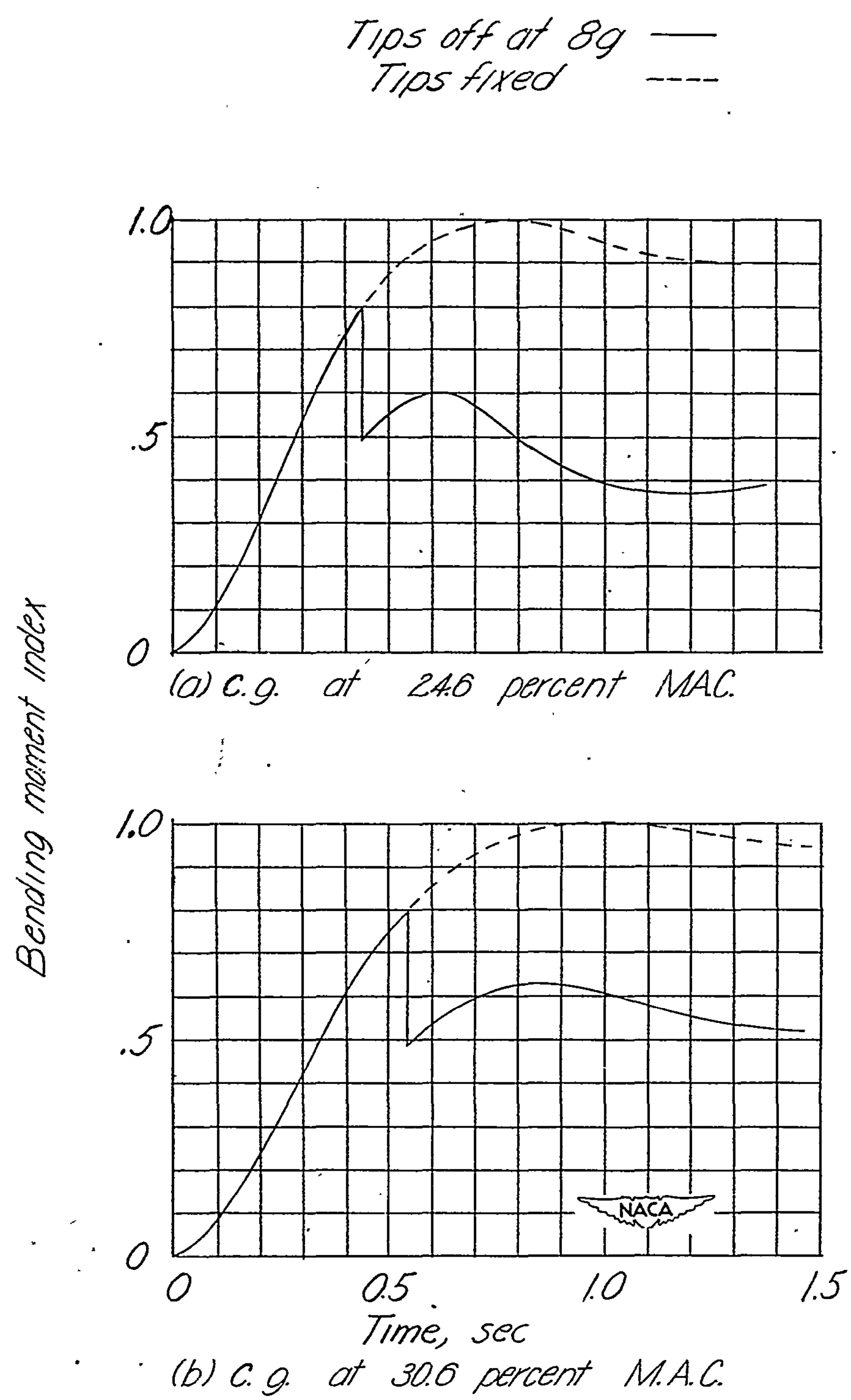



Figure 4.- Time histories of bending moment index in abrupt pull-ups for wing tips fixed and tips jettisoned at 8g.

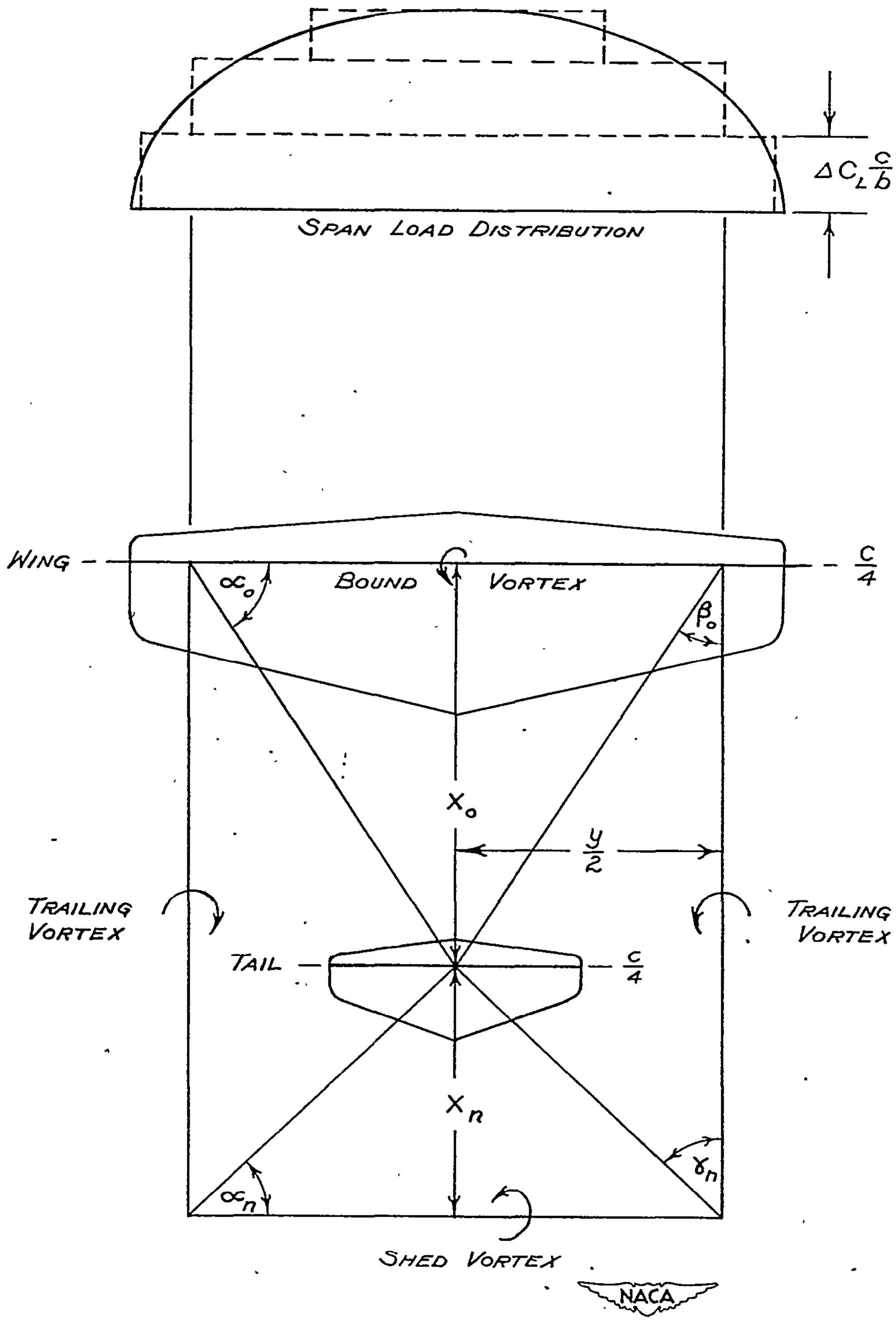



Figure 5.- Method used to calculate transient effects.