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RESEARCH MEMORANDUM

LOADS DUE TO CONTROLS AT TRANSONIC AND LOW SUPERSONIC SPEEDS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

June 7, 1957

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SUMMARY

Some results of recent experimental investigations at supersonic and transonic speeds are presented to show the present status in the estimation of load distributions on controls and adjacent wing surfaces resulting from the deflection of flap controls and spoiler controls. The results indicate that the development of methods for predicting loads associated with controls has not kept pace with the acquisition of experimental data. At low supersonic speeds sweeping the hinge line induces strong three-dimensional-flow characteristics which cannot be treated by the simplified methods previously developed for controls without sweep. At transonic speeds the estimation of loads associated with controls must usually be dependent upon experimental information inasmuch as the latest attempts to predict chordwise and spanwise loadings have met with only limited success.

INTRODUCTION

Two problems must be considered in the estimation of loads associated with controls: the direct loads on the controls themselves, and those loads induced on adjacent surfaces. In the last few years a fairly large amount of experimental information has been obtained on these types of loads. Some success in the prediction of these loads has also been realized at supersonic speeds, but at transonic speeds success in the prediction of these loads has been much more limited.

In this paper some results of the most recent investigations at supersonic speeds pertaining to flaps with a swept hinge line and to spoilers are discussed. However, most of the paper pertains to transonic results, since the transonic range is where the loads problems of most concern still appear to exist. An attempt is also made to indicate the present state of the art in the estimation of loads due to controls. The results at supersonic speeds were obtained in the Langley 4- by 4-foot supersonic pressure tunnel, and the results at transonic speeds were obtained in the Langley 16-foot transonic tunnel. In the tests at supersonic speeds transition was fixed near the leading edge by strips of carborundum grains, while in the tests at transonic speeds the Reynolds numbers were always sufficiently high to insure a turbulent boundary layer over most of the wing chord.

SYMBOLS

ΔC_p	increment in pressure coefficient due to control deflection or projection
$\Delta C_{p,R}$	resultant difference in pressure coefficients between upper and lower surfaces due to control deflection
$\dfrac{\Delta c_n c}{\Delta C_N c_{AV}}$	normalized section normal-force loading parameter due to control deflection
Δc_n	section normal-force coefficient due to control deflection
ΔC_N	wing panel normal-force coefficient due to control deflection (based on semispan wing area extended to fuselage center line)
A	aspect ratio
HL	flap hinge line
b	span
c	local chord
c_{AV}	average chord
M_∞	Mach number
x	chordwise distance
y	spanwise distance
α	angle of attack
δ_a	angle of aileron deflection
δ_d	deflector projection
δ_s	spoiler projection

$\Lambda_{c/4}$ sweep angle at quarter-chord line

Λ_{HL} sweep angle of flap hinge line

Subscripts:

s spoiler

d deflector

RESULTS AND DISCUSSION

Supersonic Speeds

At supersonic speeds simple configurations were initially studied, such as unswept-wing and delta-wing configurations, and considerable success was attained in predicting the loadings, except possibly for the high-angle-of-attack, high-control-deflection case. (See ref. 1.) Now configurations are being investigated for which more difficulty is expected in the prediction of loads; for example, a sweptback-wing model with a swept-hinge-line flap was recently investigated, and some of the results are presented in figure 1.

Effects of hinge-line sweep.- Figure 1 shows the effect of flap hinge-line sweep and wing sweep at a Mach number of 1.61 on the chordwise loading due to flap deflection. On the vertical scale of the plots in this figure are shown the changes in pressure coefficient on the upper and lower surfaces that are due to flap deflection. These values are plotted against fraction of the local wing chord. The curves in these plots represent loadings at the various spanwise stations indicated on the model sketches. The plot on the left represents the unswept-hinge-line configuration for which it was previously shown (ref. 1) that the flap load as shown behind the hinge line is about 0.7 of the value predicted by linear theory if the carryover on the wing ahead of the hinge line is not too large. It was also previously indicated in reference 1 that the results for this unswept-hinge-line case can be used for prediction of control loadings at higher Mach numbers, perhaps up to Mach numbers of 3.5 or 4.0. For this case there was very little effect of spanwise location on the flap loading or on the wing carryover load as shown ahead of the hinge line. There was no carryover load on the wing upper surface.

In figure 1 the pressure plot for the swept-hinge-line model shows that spanwise location had a considerable effect on the loading, particularly on the carryover load ahead of the hinge line. Both angle of attack and control deflection are more important factors for this configuration than for the unswept-hinge-line configuration. For the swept case the complexities of three-dimensional flow are of great concern. Also the effect of sweep may have reduced the local Mach numbers normal to the flap hinge line sufficiently to cause complications of mixed subsonic and supersonic flows such as are typical in the transonic range. At present no reliable general procedure for estimating flap or carryover loads for the swept-hinge-line case is available. There are indications that a solution at higher Mach numbers will be simpler. For example, it was shown in a paper on loads at hypersonic speeds (ref. 2) that the flow over configurations of this sweep may generally be regarded as two-dimensional.

Prediction of spoiler loads on three-dimensional wings.- In addition to the theoretical approach to the problem of loads due to controls, another approach which is sometimes useful is to apply data for simple configurations to more complex configurations. For supersonic speeds pressure data are available on a flat plate with an unswept spoiler (ref. 3) and also on a three-dimensional wing with a similar unswept spoiler (ref. 4). A preliminary evaluation of this approach was made, and the results are shown in figure 2.

The top part of figure 2 shows the wing and its spoiler, which had a height of 5 percent of the mean aerodynamic chord. The lower part of figure 2 shows ΔC_p, which represents the change in loading on the upper surface due to a spoiler deflection, plotted against chordwise distance from the spoiler in terms of spoiler height. The lower-surface pressures were not affected by spoiler deflection. The symbols represent the wing data for angles of attack of 0° and 12° which were obtained at the spanwise station indicated in the sketch of the wing. Spanwise location had no effects on these data, except near the wing tip. The solid lines represent the flat-plate data which, in this case, are two-dimensional and which were obtained at the same Mach number as the wing data. For the 12° angle-of-attack case, however, it was necessary to adjust the flat-plate data by a method developed by Lord and Czarnecki in reference 4. In simple terms, this method adjusts the data by taking into consideration the changes in Mach number on the basic wing which are due to angle of attack.

The agreement between the wing data and the adjusted flat-plate data is generally very good. In other words, it appears that results for a flat plate can be used to predict the load, including the effects of angle of attack, on a wing where the flow is approximately two-dimensional. It is also believed that there may be a possibility of obtaining agreement in cases where the flow over the wing tends to be more three-dimensional, although sufficient data to test this belief are lacking. For any case, however, the flow over the wing must not be stalled, and there must be no sharp changes in the pressure distribution on the wing without the spoiler.

Transonic Speeds

Estimation of chordwise loadings.- For transonic speeds the problems of predicting loads associated with controls are generally more difficult than for supersonic speeds. One of the most difficult problems is the prediction of the chordwise loads due to control deflection. Some idea of the present status of this problem for a typical swept-wing configuration is shown in figure 3.

In the top part of figure 3 is shown a swept-wing configuration which has an inboard flap. In the plots at the bottom of the figure, $\Delta C_{p,R}$ represents the total change in wing and flap loading due to flap deflection and is shown on the vertical scales. These coefficients are plotted against fraction of the local wing chord as the abscissa. The solid lines in these plots represent experimental pressure data which were obtained at a Mach number of 0.98 for an angle of attack of 0° and a flap deflection of -15°. The dashed line represents calculations from a lifting-surface theory for a Mach number of 1.0. The theory was very recently developed for the flap-deflected case by Keith C. Harder and E. B. Klunker of the Langley Theoretical Mechanics Division, and these unpublished results are based on the slender-wing theory by Jones (ref. 5) as extended to the case with trailing-edge sweep by Mirels (ref. 6).

This theory predicts the chordwise loading fairly well at the inboard station; however, at the outboard station the theoretically determined loading is located forward of most of the experimentally determined loading. Perhaps the main reason for this difference in load distribution is that in the theoretical case the shocks and Mach lines extend normal to the free stream, whereas in the experimental case it is known that the shocks and Mach lines are swept. Better agreement could probably be obtained for a configuration having a lower aspect ratio; however, since the theory is very new, no attempt has been made to define its range of application.

Other theoretical approaches to the problem of chordwise loading at transonic speeds such as the use of the hodograph technique are also being considered. In the meantime, some experimental loads data also have been obtained for a fairly large range of thin-wing (thickness-to-chord ratios of 0.03 to 0.06) configurations. These configurations are indicated in figure 4.

New experimental loads data.- The upper part of figure 4 shows configurations having flap controls. The configurations shown in the bottom part of the figure have spoiler controls. The sketch on the lower right shows an enlarged cross section of the spoiler-slot-deflector control. For some of these configurations only force data were obtained, including the forces and moments on the controls; however, for most of the configurations pressures were obtained over both the wings and controls. All of these configurations were investigated at angles of attack up to about 23° with generally a fairly large range of control deflection. The maximum Mach number was usually 1.03. The results for the configuration at the upper left are the only results that have so far been reported (see ref. 7).

An attempt has been made to use the data for these various configurations to develop empirical methods for estimating chordwise load distribution due to control deflection; however, little success was realized in deriving any simple correlations for even the loading over flaps. Thus, for the chordwise pressures or loadings, experience indicates that, in general, recourse must be made to data for configurations that approximate the configuration being designed.

Fortunately, for transonic speeds the prediction of spanwise load distributions due to control deflection is not always as difficult as the prediction of chordwise loadings. Studies indicate that the shape of these span-load distributions is essentially unaffected by Mach number in the transonic range for some configurations. This lack of Mach number effect suggests the possibility of using subsonic theory for prediction. Figures 5 to 7 show results of a study made of this possibility and also of the effects of Mach number for some of the configurations shown in figure 4. Most of these results are for spoiler controls since such controls are of considerable interest, and fewer loads data exist for spoiler controls than for flap controls.

Flap and spoiler at wing trailing edge.- Figure 5 which pertains to spanwise load distribution indicates the span loadings for an inboard flap configuration and an inboard spoiler configuration. The load distributions shown in the top part of the figure are due to the deflection of the inboard flap, and the load distributions shown in the bottom part of the figure are due to the deflection of the inboard trailing-edge spoiler. Both of these controls were tested on the same swept-wing—body combination. On the vertical scale is shown the weighted section normal-force coefficient due to control deflection divided by the wing-panel normal-force coefficient due to control deflection. The coefficients are plotted against fraction of wing semispan. The curves show the incremental load distribution over the wing and control, but because the curves have been normalized the actual magnitude of the load, of course, is not indicated. The experimentally determined data are indicated by the symbols and are presented for an angle of attack of 4°.

For both configurations these symbols show that increasing the Mach number from 0.80 to 0.98 had very little effect on the incremental load distribution. These results are representative of those found to exist for angles of attack from about 0° to 6°. Because the inboard end of the flap was actually slightly outboard of the fuselage, the loadings for the flap configurations show a large decrease in this region.

The solid lines in figure 5 represent calculations for a Mach number of 0.80 which are based on a subsonic theory as presented by DeYoung (ref. 8) in which the inboard flap and the inboard trailing-edge spoiler were differentially deflected. This theory was devised to apply to flap configurations as long as flow separation does not occur on the wing. Since no theory is available for the spoiler case, calculations for a flap at an arbitrary deflection and with the same span as the trailing-edge spoiler are compared with the spoiler results. For the flap configuration the calculations show very good agreement with the experimental data; however, for the spoiler configuration the agreement, as would be expected, is not as good.

For outboard controls larger effects than those shown in figure 5 can probably be expected. It has been previously shown (ref. 9) that Mach number has fairly large effects on the lateral position of the center of load due to the deflection of outboard flaps; thus, it appears that subsonic theory cannot generally be extrapolated to as high Mach numbers for the outboard-control case as for the inboard-control case.

Spoiler controls ahead of wing trailing edge.- Figure 6 shows spanwise load distribution for a wing with spoiler controls located ahead of the trailing edge. These controls were more highly swept than the spoiler control shown in figure 5 and extend over more of the wing semispan. The distributions shown in the top portion of figure 6 are due to the deflection of a flap spoiler. The distributions shown in the lower part of figure 6 are due to the deflection of a spoiler-slot-deflector on the same model. This type of control has been of particular interest because of its good effectiveness at high angles of attack.

The results of experimental investigations represented by the symbols in figure 6 show a somewhat larger effect of Mach number than is shown in figure 5. An increase in Mach number from 0.80 to 1.00 caused an outboard shift in the center of additional load for both configurations. These results are typical of results found to exist over a range of angle of attack from about 0° to 6°. The experimental span-load distributions for the spoiler-slot-deflector configuration are considerably different from those for the spoiler configuration. These differences are mainly due to the influence of the deflector on the flow. The solid lines shown here represent subsonic calculations similar to those shown in figure 5 except these calculations are the average of curves from references 8 and 10 since the controls were restricted to one wing panel. It is seen that the agreement between the calculations and experiment in figure 6 is about the same as that shown for the spoiler configuration in figure 5.

Loads at a high angle of attack.- Thus far in the discussion of spanwise load distributions at transonic speeds, the high-angle-of-attack case has not been considered. In figure 7, therefore, are shown distributions at a high angle of attack for the spoiler-slot-deflector configuration shown in figure 6. Subsonic calculations for a Mach number of 0.60 are indicated by the solid line. The experimental data are indicated by the dashed lines.

The results for this particular configuration indicate that Mach number has a large effect and that theory no longer predicts the shape of the loadings. These effects are mainly the result of flow separation in the region of the wing tip; therefore, as is well known, when separated flow has a large effect on loading then recourse must be made to experimental data for loads information.

Loads on control surfaces.- Now that the discussion in this paper on the recent overall wing-loads information has been concluded, the control loads on the flap spoiler and the spoiler-slot-deflector are considered. Some of these control loads are shown in figure 8.

In the lower part of figure 8 is presented a comparison of the spanwise load distribution for the spoiler and deflector components of these controls. On the vertical scale is plotted the weighted control section normal-force coefficient divided by the total control normal-force coefficient. These coefficients are plotted against fraction of the wing semispan. The span-load distributions are for an angle of attack of 0° and a Mach number of 1.00. Other unpublished data for the same configurations show very little effect of Mach number for Mach numbers from 0.60 to 1.03. The results show that the type of control component has very little effect on the distributions; therefore, these distributions appear to be primarily affected by the wing geometry and the control location.

In the upper right part of figure 8 is shown the variation of total control normal-force coefficient with angle of attack for the spoiler and deflector components of the spoiler-slot-deflector. The solid and dashed lines indicate the effect of increasing Mach number from 0.60 to 1.00. For both the spoiler and deflector components this Mach number effect is not very large over the range of angle of attack. The normal-force coefficients for the spoiler portion of the control decrease in magnitude with increasing angle of attack, whereas those for the deflector are not affected. The data in the present case and other available data (ref. 11) indicate that for large control deflections tests need be made only at low angles of attack in order to establish the maximum control loads.

CONCLUDING REMARKS

The results of this study show that success in predicting loads associated with controls has not kept pace with the acquisition of experimental data. At supersonic speeds data for loads are now available for swept-flap configurations, but methods of prediction for these cases must still be developed. At transonic speeds more experimental data are available, but only limited success in correlating experimental data with theory has been realized.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., March 5, 1957.

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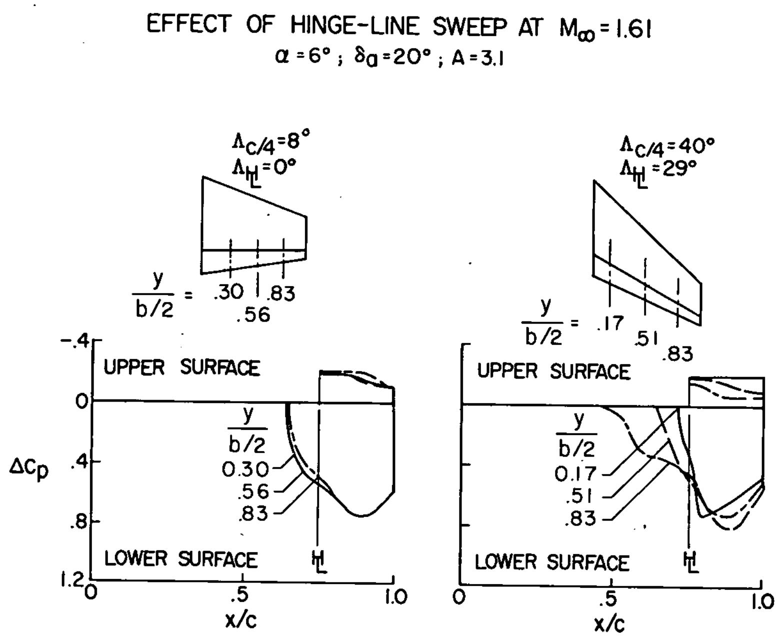

Figure 1

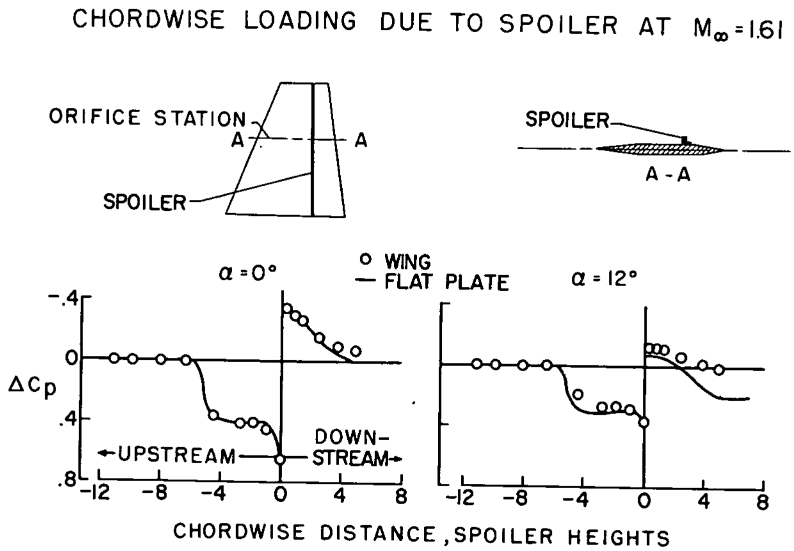

Figure 2

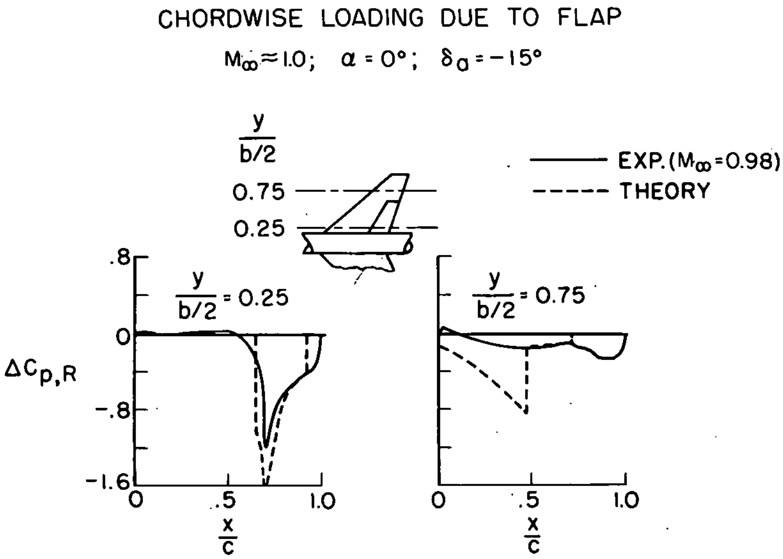

Figure 3

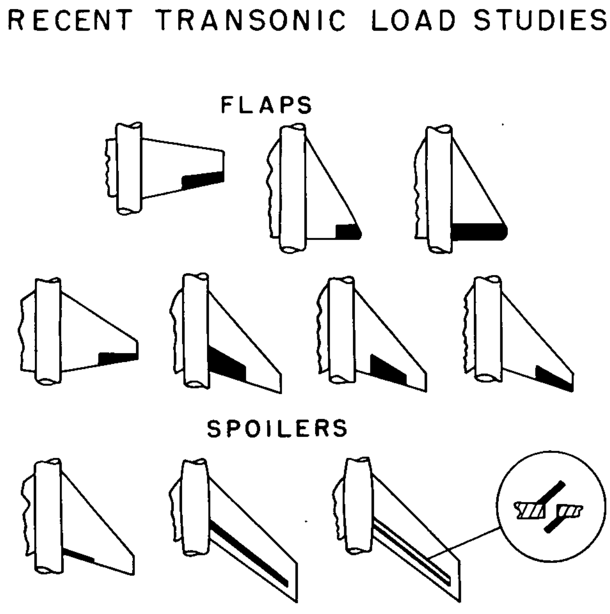

Figure 4

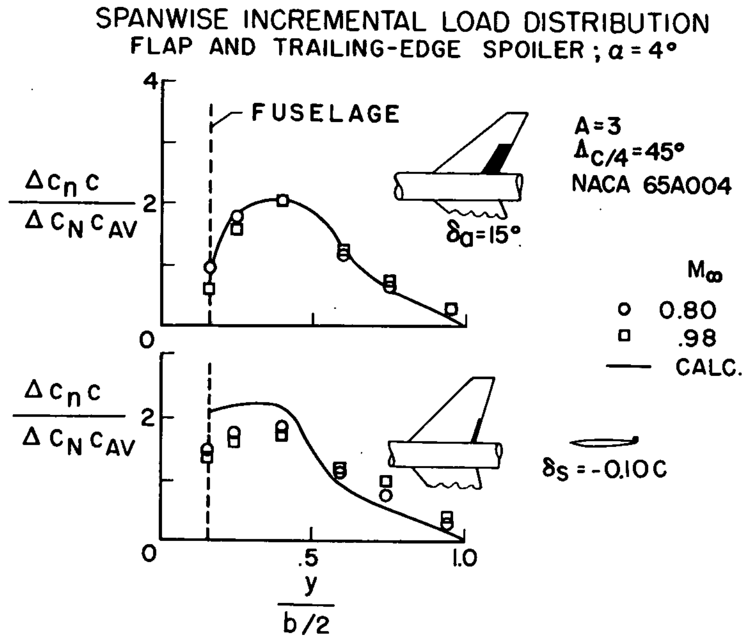

Figure 5

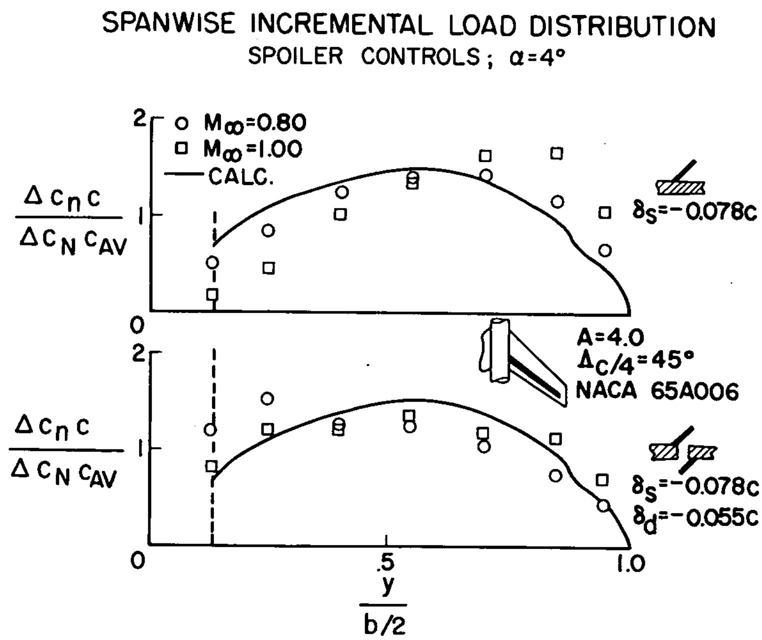

Figure 6

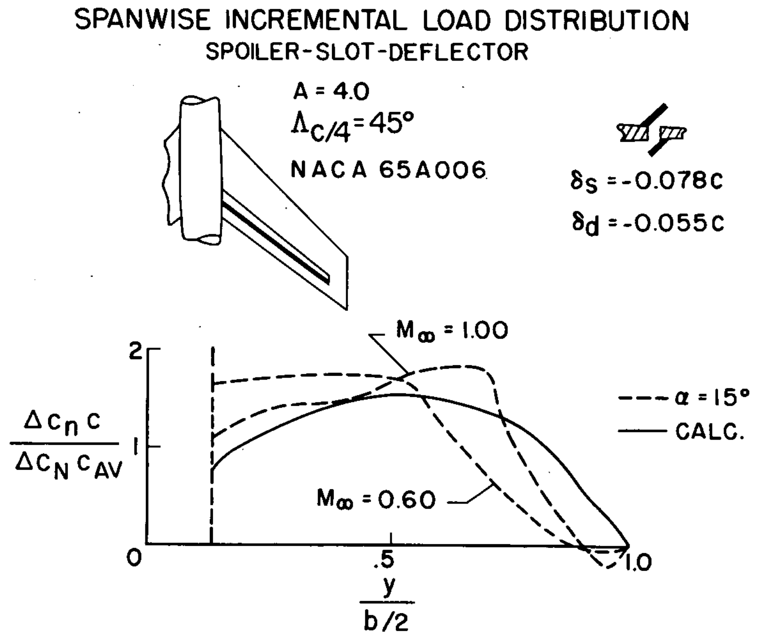

Figure 7

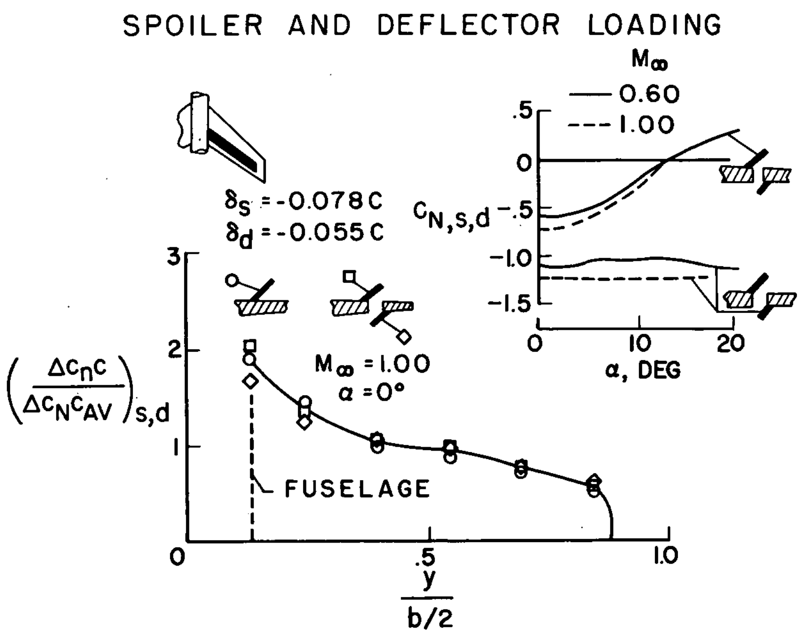

Figure 8

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