ICAS-82-5.8.4

ANALYSIS OF JET TRANSPORT WINGS WITH DEFLECTED CONTROL SURFACES BY USING A COMBINATION OF 2- AND 3-D METHODS

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ABSTRACT

A computational technique has been developed which analyzes transport wings with deflected spoilers or ailerons. It uses 2-D separated flow analysis results in a "strip theory" fashion to provide inputs for a 3-D lifting surface method. Wing body lift and pitching moments are quantitatively predicted and calculated spanloads qualitatively match experimental results. The method is accurate and economical enough to be useful in the basic design of control systems.

INTRODUCTION

Stability and control engineers are often faced with analyzing swept wings with deflected ailerons or spoilers. Design of such surfaces has been based heavily on empirical techniques such as those in the DATCOM [1]. Once a trial configuration has been estabished, extensive amounts of wind tunnel testing are usually undertaken to modify and verify the configuration. Since testing is usually not done at flight Reynolds' number, significant uncertainty may exist until flight testing begins. At this stage it can be very costly to make design revisions. For this reason, the development of computational techniques to model these separated flows and allow quick comparisons of various configurations is highly desirable. This would allow for a more intelligent use of valuable wind tunnel time, preferably at flight Reynolds' number. Together this should lead to fewer changes and better performance at the flight test stage.

The effects of control surface deflections on swept wing aircraft push the prediction capabilities of the usual inventory of aerodynamic computer codes to their limits. As long as the flow remains attached, reasonable estimates of aerodynamic parameters may be obtained. However, when ailerons or spoilers induce separation over even a small portion of the wing, the usual 3-D flow analysis programs are of little use.

As an alternative to these standard codes, a method has been developed which utilizes the combination of a 2-D separated flow analysis and a 3-D lifting surface program. A combined method of this type may not appear as fundamentally valid as a complete three-dimensional, viscous, separated flow program. However, even if such an all encompassing program were available, it would be costly to execute on a regular basis. Therefore, assuming quantitative results of acceptable uncertainty are possible with this combination of two and three-dimensional analyses, it would prove to be of great value in the regular preliminary design calculations of an aircraft.

FLOW DESCRIPTION

The deflection of a spoiler on a wing produces separated flow for all but the smallest deflection angles. Similar separations are caused by ailerons at moderate and larger deflections. For ailerons, the disturbance typically initiates along the hingeline. For spoilers, the region ahead of the deflected surface may show a disturbance in the form of a separation-reattachment bubble parallel to the hingeline. This is in addition to the separation which starts at the spoiler trailing edge and continues past the trailing edge of the wing.

Figure 1 represents china clay surface flow patterns obtained experimentally for a typical transport wing with a spoiler. This figure shows that the flow is well behaved except in the immediate region around the spoiler. Ahead of the spoiler the flow starts out generally in the streamwise direction. As the surface flow slows, the spanwise velocity component dominates and the flow lines (except where very close to the inboard edge of the spoiler) turn outward in the spanwise direction. The flow just above the surface does not turn nearly so much since the streamwise velocity in the outer portion of the boundary layer does not slow to the same degree. This allows the main flow to leave the surface of the airfoil in a more nearly streamwise direction and reattach to the face of the projected spoiler. This reattachment is indicated by a spanwise attachment line halfway up the spoiler surface. The region between the separation and reattachment lines shows the surface pattern for a hingeline vortex with considerable spanwise flow. The surface flow near the ends of the spoiler is characterized by the spilling of the air past the exposed edges. The region behind the spoiler shows no distinct surface flow pattern.

An experimental survey of total pressure in the wake behind a wing body with nacelle and spoiler is shown in Figure 2. The survey was made perpendicular to the stream, slightly behind the wing tip. The boundary layer over a portion of the fuselage is visible on the right side of the figure. Nacelle pressure losses

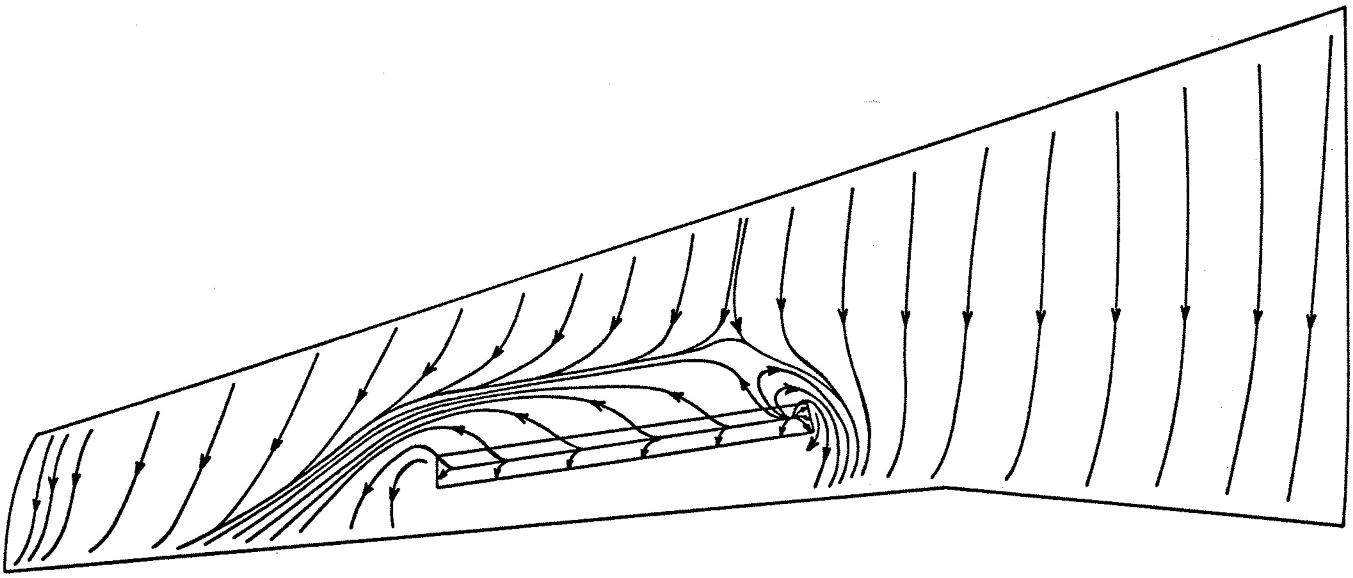

FIGURE 1 CHINA CLAY SURFACE FLOW PATTERNS FOR A WING WITH A SPOILER

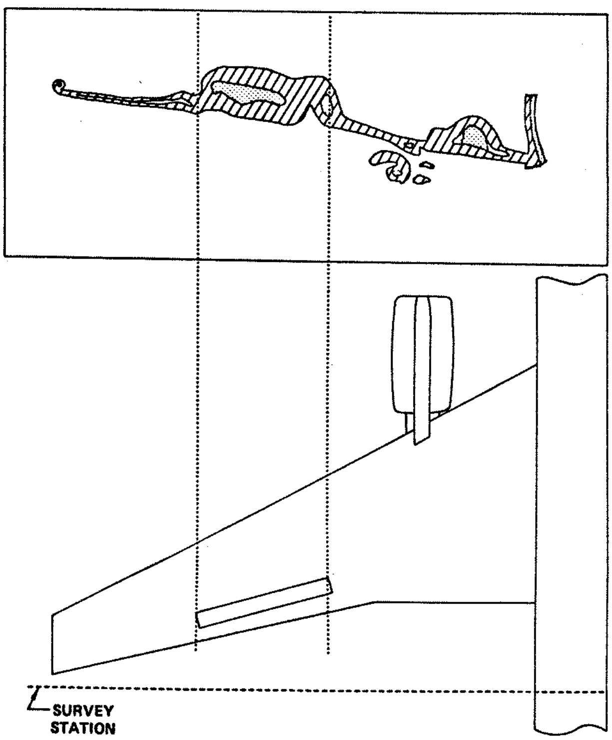

FIGURE 2 WAKE IMAGING SYSTEM SURVEY OF A WING BODY WITH SPOILER AND NACELLE

are visible below the wing as are nacelle strut losses on top of the inboard portion of the wing. (The size of the strut losses are due to a less than optimal strut design for this model.) The wing tip vortex is apparent at the far left and the effect due to the boundary layer can be seen along the wing. The region immediately behind the spoiler shows massive loss in stagnation pressure.

The effect of the spoiler on the wake pressure is highly localized to the region immediately behind the spoiler. Similarly, the effect of the spoiler on the surface flow pattern is limited to the immediate vicinity around the spoiler.

ANALYSIS TECHNIQUE DEVELOPMENT

Both the 2-D and 3-D computation methods, in their present forms, are not designed to treat the full problem. The current 3-D methods do not address the problem of wings with part span control surface separations. Some 2-D programs have been developed which handle this type of separation [2, 3, 4], but they are clearly limited by their 2-D restriction. However, through the judicious merging of 2-D and 3-D methods, it is possible to produce a very useful and economical solution technique.

A system that correlates 2-D and 3-D methods in the design of high-lift configurations was suggested in Reference 5. Sections were analyzed at selected spanwise stations using a 2-D viscous-potential code [2]. Using these section results, camberlines were defined at each section for use in a 3-D lifting surface program [6]. Some separation on trailing edge flaps was modeled by reducing the flap angle for the lifting surface program when the 2-D code indicated separation. Wing lift, pitching moments, and spanloads were well predicted for high-lift configurations, even when separation was present on a trailing edge flap.

The deflected control surface cases are quite similar to the separated trailing edge flap case. In either case the disturbance is localized to the immediate region around the disturbing surface. This causes abrupt variations in local section lift along the span. In the flap case, these changes are modeled by the absence or presence of a flap camber sheet along the span. For the deflected control case, simple addition of a secondary camber sheet is not reasonable. Instead, an effective camber sheet must be established for the wing in the control surface region. In order to correctly specify the camber lines here and along the rest of the span, 2-D viscous-potential results are used.

In the case of a clean wing with attached flow, a first approximation of the camber sheet is the local geometric camberline at each section along the span. The difference in boundary layer thickness between upper and lower surfaces becomes significant as the angle of attack is increased. This causes a decrease in the effective camber of a given section (Figure 3).

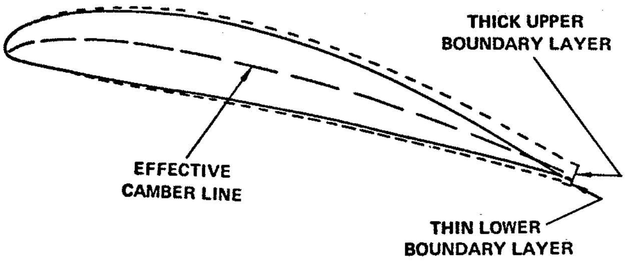

FIGURE 3 BOUNDARY LAYER DECAMBERING

Similar decambering is noted at lower angles of attack for sections with deflected spoilers or ailerons (Figures 4 and 5). The major decambering in these cases, however, is due to the separated wake behind the control surface. The 2-D viscous-potential code enables the calculation of these effective camberlines.

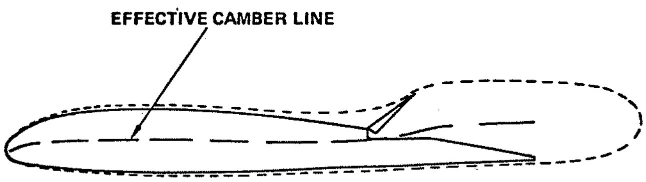

FIGURE 4 DECAMBERING DUE TO A DEFLECTED SPOILER

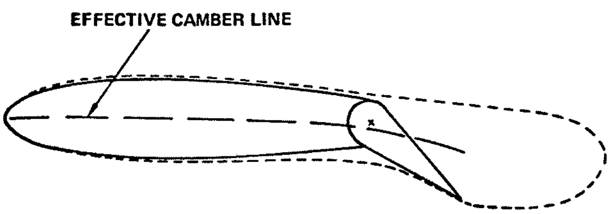

FIGURE 5 DECAMBERING DUE TO A DEFLECTED AILERON

Camberlines are calculated, by the 2-D code, along the span in "strip theory" fashion and assembled to produce a camber surface as shown in Figure 6. The camber surface is then used as input for the 3-D lifting surface program. The final output is in the form of lift, pitching moment, and spanload results.

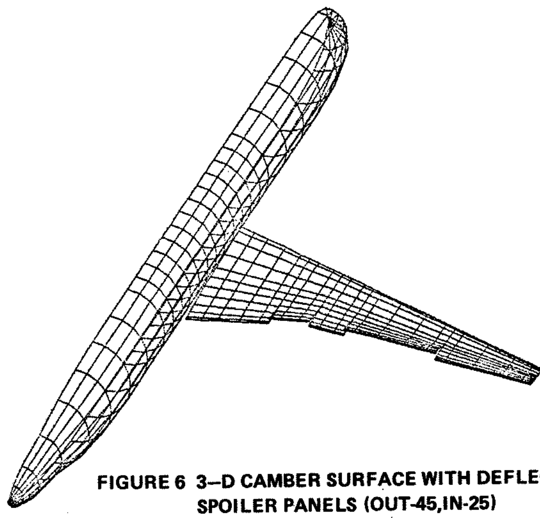

FIGURE 6 3-D CAMBER SURFACE WITH DEFLECTED SPOILER PANELS (OUT-45,IN-25)

RESULTS AND DISCUSSIONS

The flow past a transport wing body combination with and without deflected inner and outer spoiler panels, was analyzed for various angles of attack at Mach 0.7. The spoiler deflections used were $20^\circ/30^\circ$ (inner/outer) and $25^\circ/45^\circ$. Figure 6 shows the effective camber surface for the $25^\circ/45^\circ$ case. A comparison, of predicted with experimental spanloads, is presented in Figure 7 for this configuration. Comparisons of calculated with experimental lift and pitching moment for the $20^\circ/30^\circ$ case are shown in Figures 8 and 9.

It should be noted that the wind tunnel model used for the experimental results had a nacelle mounted at ~ 0.3 span. The computations were made with no nacelle, because it was felt that the nacelle modeling in the 3-D program has not been fully developed. From other experimental testing, it was observed that the nacelle effect on total wing lift is very small over a wide range of angles of attack.

The nacelle does cause slight variations in the local spanload. For the linear region of the lift curve, the experimental effect of the nacelle can produce a small change in pitching moment slope. Therefore, the comparisons between experimental and computed results should not be exact, but should represent a reasonable match.

The spanloads in Figure 7 were calculated at angles of attack which produced roughly the same integrated wing lift as was measured experimentally. This caused only a slight variation from the experimental angles of attack. The spanload results show very good agreement between experiment and computations for the spoiler nested case. The presence of the nacelle is apparent as a mild ripple in the experimental results. The spoiler deflected case shows that the computation technique reasonably models the spanload trends. The quantitative agreement is not as good, but the general trends are well described.

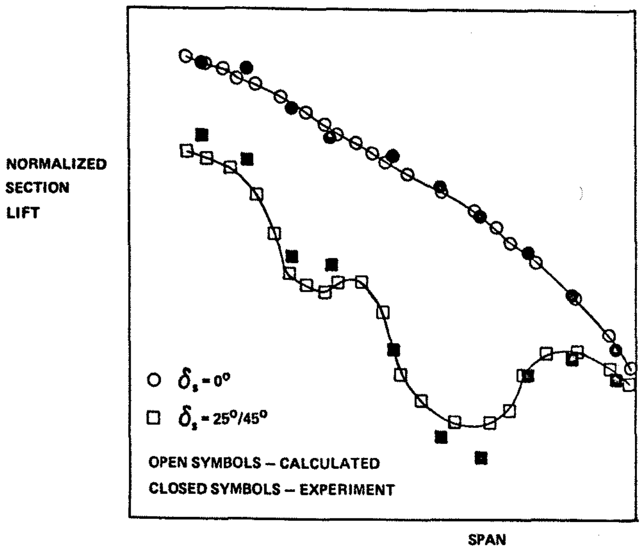

FIGURE 7 COMPARISON OF CALCULATED AND EXPERIMENTAL SPANLOADS FOR $\delta_s = 0^\circ$ AND $\delta_s = 25^\circ/45^\circ$

The lift comparisons in Figure 8 show fair to excellent agreement within the linear region of the curve. Similar agreement is shown for the pitching moment results in Figure 9. The discrepancies noted in these plots could be due to several factors. Foremost would be the nacelle difference described above. Another likely factor is the simple modeling of the Mach number effect. Figure 10 shows that the experimental Mach number effects are significant for this configuration.

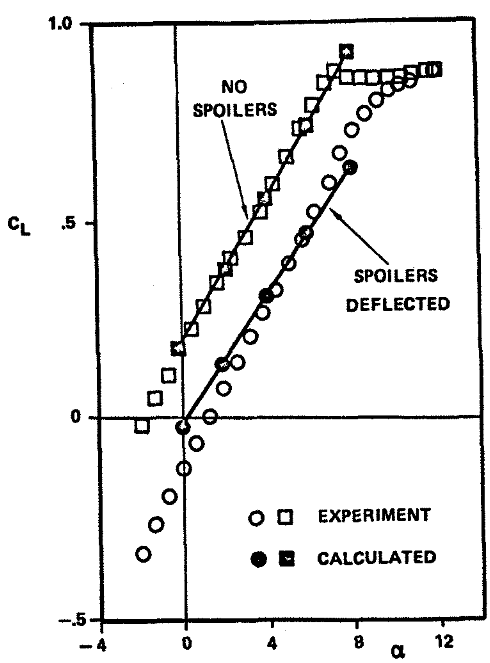

FIGURE 8 COMPARISON OF CALCULATED & EXPERIMENTAL LIFT

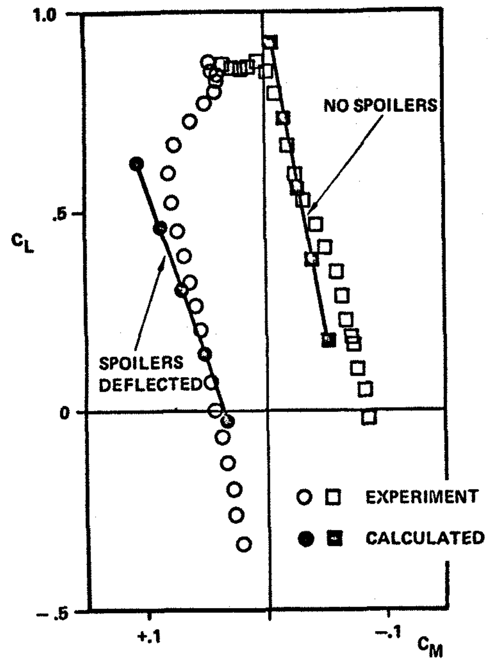

FIGURE 9 COMPARISON OF CALCULATED & EXPERIMENTAL PITCHING MOMENT

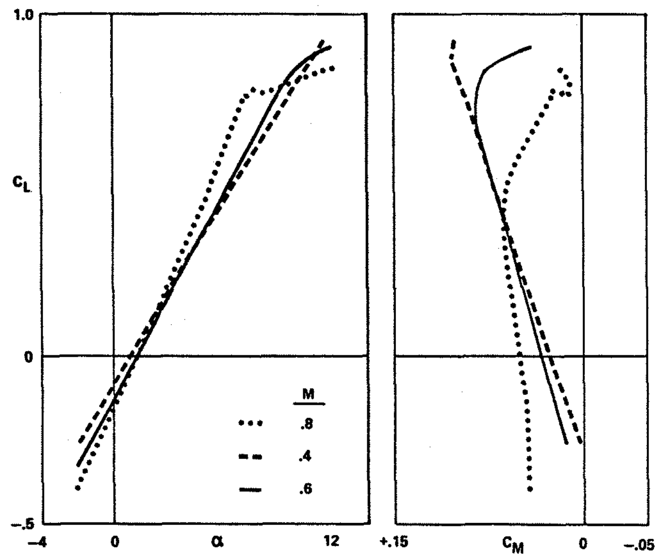

FIGURE 10 EXPERIMENTAL RESULTS SHOWING THE MACH NUMBER EFFECT ON LIFT AND PITCHING MOMENT

Drag values computed by the 3-D program have not been presented. The thin camber sheet modeling of the wing does not yield reasonable drag values for the separated cases. Empirical methods prevail here, although an integration of 2-D results may provide useful insight.

Regardless of these small deviations between calculated and experimental results, the general trends for the wing body forces, moments, and wing spanloads are well predicted. A major value of this technique is the ability to make quick comparisons of various cases of spoiler deflections. For these comparisons, any of the discrepancies noted above would be even less important, because they would not greatly affect the assessment of relative merit for a given spoiler case.

CONCLUSIONS

1) Wing body lift and pitching moments have been well predicted for deflected control cases.

2) Calculated spanwise lift distributions match the general trend of deflected control experimental data.

3) Calculated drag is grossly underestimated due to the simplicity of the model.

4) This method is useful for wings with spoilers and ailerons. Separations caused by these devices tend to have well-behaved separation lines and are present only near the trailing edge.

5) The suitability of this method is unclear for high angles of attack. Part span leading edge vortex separations are present on swept wings at high incidence angles. The "strip theory" approach may not be valid for these massive vortex separations.

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