**Table 1 Pressure integration summary for Mach 10 solutions**

| Part/solution | Lift, N/unit span | Thrust, N/unit span | Pitching moment, N-m/unit span |
|---|---|---|---|
| Forebody/PNS | 16.9949 | −3.8063 | −3.4412 |
| Forebody/RANS | 16.6635 | −3.6400 | −3.3562 |
| % Difference | −1.9 | −4.4 | −2.5 |
| Cowl/PNS | 0.7086 | 0.0000 | −0.0335 |
| Cowl/RANS | 1.0155 | 0.0000 | −0.0295 |
| % Difference | +43.3 | 0.0 | −11.9 |
| Aftbody/PNS | 32.5650 | 8.9916 | 4.6296 |
| Aftbody/RANS | 32.3684 | 8.9516 | 4.5823 |
| % Difference | −0.6 | −0.4 | −1.0 |

**Table 2 Total lower surface pressure integration for Mach 10 solutions**

| Total lower surface integrated pressures | Lift, lb/unit span | Thrust, lb/unit span | Pitching moment, in.-lb/unit span |
|---|---|---|---|
| With cowl separation | 49.9371 | 5.3517 | 1.2399 |
| Without cowl separation | 50.0474 | 5.3116 | 1.1965 |
| % Difference | +0.2 | −0.7 | −3.5 |

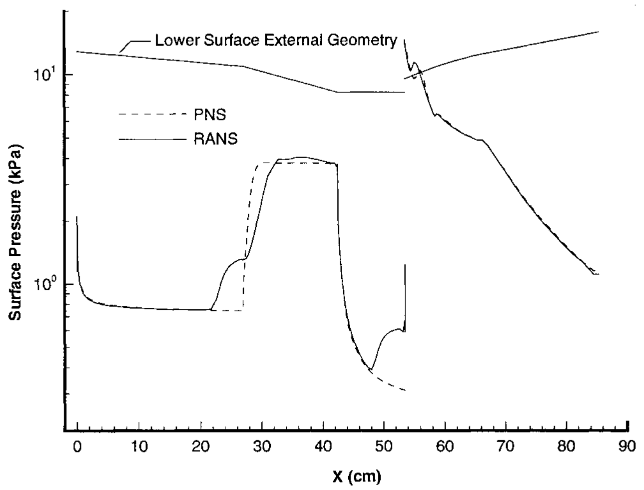


**Fig. 3 Lower surface pressure comparison, PNS vs RANS solutions, $M_\infty = 10.0$, $Re_\infty = 6.6 \times 10^6$/m, NPR = 4000.**

Another concern about plume-induced flow separation is that high-temperature exhaust gas may actually be propagating upstream, causing increased thermal stress on the affected parts of the cowl. Plots of exhaust-gas mass fractions near the cowl trailing edge showed that the plume doesn't truly propagate upstream. Thus, it would not be expected that the external part of the cowl will require special cooling or material requirements to sustain the plume-induced separation behavior.

## Conclusions

This study attempted to provide a better understanding of the possible existence and effects of plume-induced flow separation on the cowl of a hypersonic airbreathing vehicle employing scramjet exhaust flow simulation. For representative conditions at a freestream Mach number of 10, the results showed that flow separation was predicted on a generic configuration and resulted in a negligible difference in lift and thrust, while there was a small increase in pitching moment. For this configuration incorporating a flat cowl, no change in drag (negative thrust) was seen, but the results are likely geometry-dependent. If, e.g., the cowl trailing edge was at a position such that part of the cowl was an expansion surface, the separation would likely be greater, as would the impact on lift and thrust. It is further expected that the nature of the flow near the cowl trailing edge will preclude the plume from propagating forward and causing a region of thermal stress there.

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# Effect of Leeward Flow Dividers on the Wing Rock of a Delta Wing

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## I. Introduction

ONE of the limitations to combat effectiveness for all fighter aircraft is the phenomenon of wing rock. Attempts have been made in recent years on specific aircraft configurations[1,2] using a variety of both controls-oriented and aerodynamic fixes to control wing rock problems. Research efforts have been ongoing to understand the complexity of high-angle-of-attack vortex flows. Some examples of relatively recent studies aimed at understanding flows relating to wing rock are shown in Refs. 3–8. A significant number of these studies made use of the delta wing configuration.

Some of the aerodynamic phenomena that contribute to the wing rock of delta wings may play a part in the wing rock of a complete aircraft configuration. Of particular interest is the phenomenon of vortex position and breakdown asymmetry. During wing rock, large flow asymmetries are encountered on both the windward and leeward sides due to the changing sideslip. Large vortex asymmetries can be produced under this situation, thereby contributing to the building-up of wing rock amplitude.

Experimental studies on wing rock are inevitably affected by factors such as the particular model configuration, attachments of sensors and instrument, and model support arrangement. The specific technical objective of the present experimental study is to investigate the effect on wing rock of a symmetrically placed obstacle that deflects the leeward flow during sideslip conditions. The study was conducted on an 80-deg-sweep, sharp-edge delta wing because of the wealth of

Received Oct. 6, 1993; revision received March 9, 1994; accepted for publication March 11, 1994. 

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data available on the configuration. Deflection of the leeward reattached flow is provided by small flat plate "dividers" on the leeward surface.

## II. Approach

The experiment was performed in the University of Toledo 0.91m × 0.91m Subsonic Wind Tunnel. Tests were conducted at tunnel dynamic pressure ranging from 0.14 to 0.48 kPa, corresponding to chord Reynolds numbers ranging from 3.9 × $10^5$ to 7.1 × $10^5$.

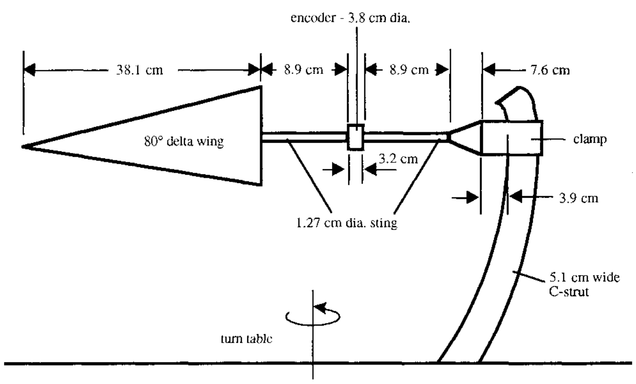
The wind-tunnel model, shown in Fig. 1a, was an 80-deg-sweep, sharp-edge delta wing. The model was 38.1 cm long and was constructed of 0.32-cm thick aluminum plate. Two low-friction bearings attached to the model sting allow the model to rotate "freely" around its longitudinal axis. An angular transducer is used to measure the roll angle history. The flow dividers, shown schematically in Fig. 1b, were constructed of thin sheet metal. Although both triangular and rectangular dividers were tested, this Note will focus only on the rectangular dividers.

Static force and moment measurements were carried out using a five-component (no axial force) strain-gauge moment balance attached to the end of the model. The balance was so located to minimize potential interference from a balance housing on the flow directly over the model. Smoke and surface flow visualizations were also performed.

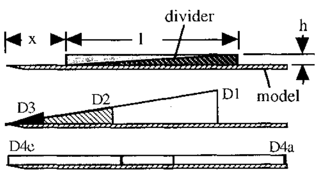
## III. Results and Discussion

### A. Wing Rock Amplitude and Frequency

Wing rock of the baseline model starts at $\alpha$ = 24 deg, and terminates at $\alpha$ = 50 deg. The typical mean roll angle is less than 4 deg. Figure 2 shows the effects of a rectangular divider at different chordwise positions. Except for D4b, changing the axial location seems to shift the initial angle of attack for wing rock, but has relatively little effect on the span of angle over which wing rock occurs. That is, a change in divider position that causes a delay in the initiation of wing rock also causes wing rock to terminate at a higher angle of attack. Except for D4e, the dividers have a destabilizing effect at lower angles of attack compared with the baseline. Overall, the result shows that the divider provides more damping at higher angles of attack when it is placed near the trailing edge, but is more destabilizing at lower angles of attack at the same placement. The effect of D4e, which starts very near the apex, is confined mainly to higher angles of attack. There seems to be a chordwise position, D4b, where the divider geometry tested produces the smallest angle span where wing rock occurs.




a)




| Label | l/c | h/b | Shape | x/c |
|---|---|---|---|---|
| D4a | 0.58 | 0.10 | rectangle | 0.41 |
| D4b | 0.58 | 0.10 | rectangle | 0.33 |
| D4c | 0.58 | 0.10 | rectangle | 0.25 |
| D4d | 0.58 | 0.10 | rectangle | 0.14 |
| D4e | 0.58 | 0.10 | rectangle | 0.01 |

b)

**Fig. 1 a) Schematic of the model setup and b) rectangular divider geometries and designations.**

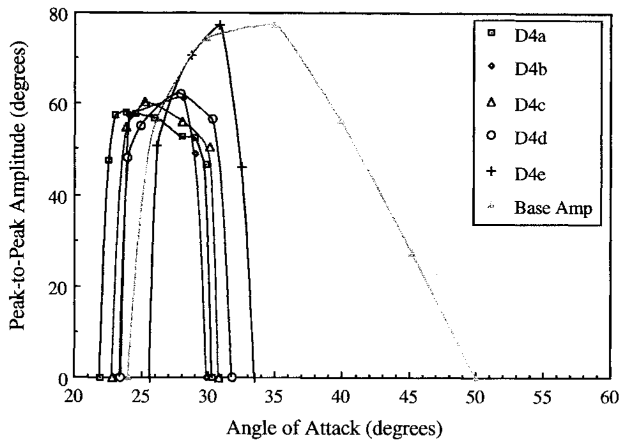



**Fig. 2 Effect of divider position on amplitude.**

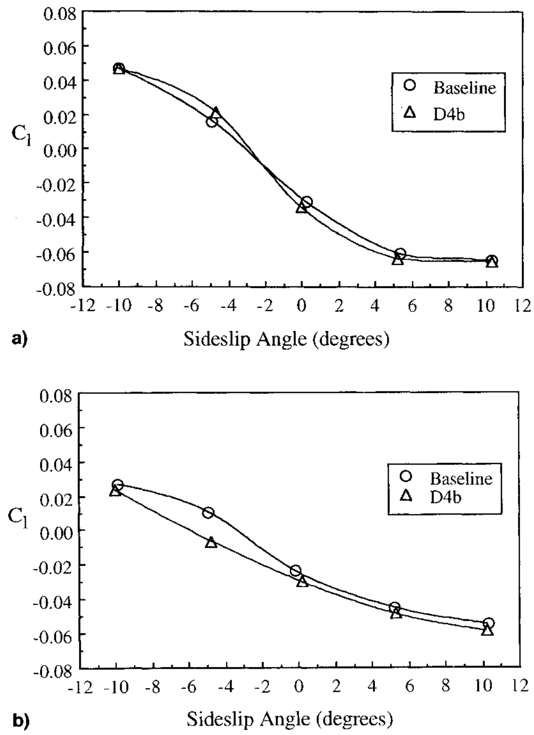
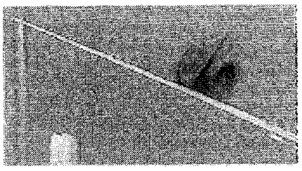


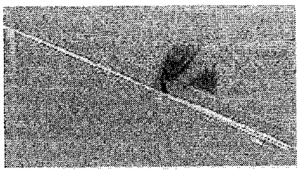

**Fig. 3 Effect of divider D4b on $C_l$ at sideslip. $\alpha$ = a) 25 and b) 35 deg.**

### B. Rolling Moment and Flow Visualization

Figures 3a and 3b show the effect of divider D4b on the rolling moment as a function of $\alpha$ at different sideslip angles $\beta$. A zero-sideslip asymmetry exists on the baseline case. The divider in general altered the asymmetry at zero yaw, but not significantly. At nonzero sideslip conditions and below $\alpha$ of about 25 deg, the divider has relatively small effect on the rolling moment. At $\alpha$ = 25 deg, shown in Fig. 3a, the divider

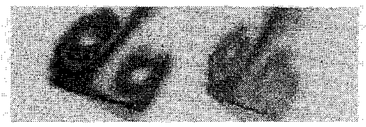


Baseline



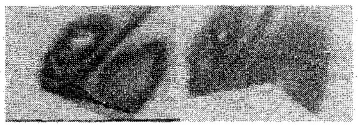
With divider

**Fig. 4 Effect of D4b on vortex positions at station 3 at $\alpha$ = 25 deg and $\beta$ = 10 deg.**



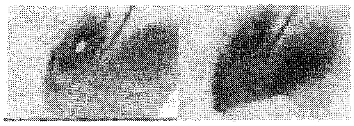
Baseline With divider

Laser sheet at x/c = 0.2



Baseline With divider

Laser sheet at x/c = 0.4



Baseline With divider

Laser sheet at x/c = 0.6

**Fig. 5 Effect of D4b at $\alpha$ = 35 deg and $\beta$ = 10 deg.**

causes a small increase in $C_l$ at small $\beta$ compared with the baseline. Correspondingly, there is a small increase in the $C_l$ gradient about zero $\beta$. An inspection of Fig. 2 shows a small increase in the wing rock amplitude at this condition.

Figure 4 shows the laser cross sections at $\alpha$ = 25 deg, and $\beta$ = 10 deg. At this condition, the divider has a noticeable effect on vortex position asymmetry. The right (windward) vortex is closer to the surface compared with the baseline, whereas the left (leeward) vortex is farther away. The flow visualization result agrees with the rolling moment result where the rolling moment at sideslip is increased by the divider.

At $\alpha$ = 35 deg, shown in Fig. 3b, the rolling moment at sideslip is reduced by the divider. The laser cross sections at $\alpha$ = 35 deg and $\beta$ = 10 deg, shown in Fig. 5, demonstrate the reason for the reduction in rolling moment. The breakdown of the windward vortex has propagated to a more forward position for the wing with the divider. The leeward vortex is displaced upward from the surface and seemingly weaker. An inspection of Fig. 2 shows a complete elimination of wing rock under this situation. Hence, the results indicate the divider decreases the static roll stability at high angles of attack, but increases the dynamic roll stability. Eventually, this leads to the suppression of wing rock.

## IV. Summary and Conclusions

The effects of a flow divider placed on the leeward side of an 80-deg sharp-edged delta wing were studied. Effects of divider geometry, sizes, and placement were investigated. With a divider in the appropriate positions, wing rock is suppressed for angles of attack above 30 deg. At the lower range of the $\alpha$, where the wing is naturally susceptible to wing rock, however, the divider can actually promote wing rock. These opposing effects on wing rock would prevent the fixed divider concept to be used for wing rock suppression.

Measurements indicate that the divider increases the rolling moment at sideslip condition for the angle-of-attack range where wing rock is promoted by the divider. Flow visualization shows that this is due to an increase in the vortex position asymmetry. The static stability is increased moderately, but the dynamic stability is reduced correspondingly. The divider therefore enhances the tendency for wing rock. At higher angles of attack where wing rock is suppressed by the divider, the rolling moment at sideslip is reduced by the divider. Flow visualization shows that this is due to the promotion of breakdown of the windward vortex. This in-turn leads to a decrease in the static stability and an increase in the damping. Wing rock is therefore suppressed.

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# Integration of the Supersonic Kernel Function

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## Nomenclature

$B$ = $\sqrt{M^2 - 1}$
$D_{rs}$ = velocity influence coefficient

Received Sept. 8, 1993; revision received Feb. 24, 1994; accepted for publication March 4, 1994. 

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