Effects of Delta Wing Separation on Shuttle Dynamics

J. PETER REDING* AND LARS E. ERICSSON†
Lockheed Missiles & Space Company Inc., Sunnyvale, Calif.

The unsteady aerodynamics of a candidate delta-wing shuttle orbiter have been investigated. Three potentially dangerous flow separation phenomena involving the delta wing have been identified, each of which could seriously compromise the flight dynamics. They are 1) leeside shock-induced separation, 2) sudden leading-edge stall, and 3) the subsonic leading-edge vortex and vortex burst. Each of these unsteady flow phenomena can be triggered by control surface deflection. Furthermore, wing stall and control-induced separation effects interact with the relatively large fuselage, increasing the coupling between lateral and directional stability characteristics. Trajectory shaping may be the most powerful means of dealing with these flow separation effects. The re-entry trajectory can be tailored to avoid or quickly traverse the unstable flow regions. However, it is prudent to use available means to control the duration and extent of separated flow when traversing the critical flow region. The unstable flow boundaries may be altered by modification of the wing planform, airfoil section, control deflections, etc.

Nomenclature

b = span
c = root chord
C_p = pressure coefficient, $(p - p_\infty)/(\rho U^2/2)$
l = rolling moment, coefficient, $C_l = l/(\rho U^2/2)(Sb/2)$
$l_\perp$ = average location of wing separation relative to leading edge
M = Mach number
m = pitching moment, coefficient; $C_m = m/(\rho U^2/2)Sc$
N = normal force, coefficient; $C_N = N/(\rho U^2/2)S$
p = roll rate
R_c = freestream Reynolds number
S = reference wing area
t = time
U = freestream velocity
$U_\perp$ = velocity component normal to wing leading edge at $\alpha = 0$, $U_\perp = U \cos \Lambda$
y = spanwise coordinate
α = angle of attack
β = angle of sideslip
Δ = increment
δ = elevon deflection angle
Λ = sweep angle
ρ = density
ϕ = roll angle

Subscripts

$b.l.$ = boundary layer
s = separated flow
w = wake

Superscripts

i = denotes separation-induced increment

Derivatives

$C_{l\beta} = \partial C_l/\partial\beta$
$C_{lp} = \partial C_l/\partial(pb/2U)$
$\dot{\beta} = \partial\beta/\partial t$
$C_{l\phi} = \partial C_l/\partial\phi$
$C_{l\dot{\phi}} = \partial C_l/\partial(\dot{\phi}b/2U)$
$\dot{\phi} = \partial\phi/\partial t$

Presented as paper 72–976 at the AIAA Atmospheric Flight Mechanics Conference, Palo Alto, Calif., September 11–13,1972; submitted September 14, 1972; revision received February 28, 1973. The results reported herein were obtained in a study conducted for the NASA Manned Spacecraft Center, Contract NAS 9-11445, under the direction of J. C. Young.

Index categories: Entry Vehicle Dynamics and Control; Jets, Wakes, and Viscid-Inviscid Flow Interactions; Nonsteady Aerodynamics.

* Research Specialist. Member AIAA.
† Senior Staff Engineer. Associate Fellow AIAA.

Introduction

THE re-entering space shuttle orbiter (referred to hereafter simply as "the shuttle") presents the dynamicist with a challenging set of flight conditions. The shuttle traverses Mach numbers from hypersonic to subsonic and angles of attack from as high as 50° down to nearly zero (Fig. 1). Generally, the envelope of flight conditions considered for the shuttle involves entry at a high initial angle of attack (in the vicinity of 30° to 50°). At $M \approx 20$ a pitchover down to between 30° and 20° is accomplished where the bank angle program is initiated to achieve the desired crossrange. This continues down to the Mach number range $7.0 \geq M \geq 4.0$, where a gradual pitchover to the cruise angle of attack ($5° \leq \alpha \leq 10°$) occurs. This last pitchover may be extended down to Mach 2.0.

A very important part of the entry trajectory is flown at high angle of attack and high Mach number. This high α-M regime is unique to the shuttle and is, therefore, the least understood. Thus, a major portion of this paper is devoted to a discussion of the shock-induced separation that occurs on the leeside of the delta wing at high angles of attack and Mach numbers. The dynamic implications of some better known delta wing flows are also discussed; more specifically, the sudden stall that can occur at low supersonic speeds when the Mach number normal to the leading edge is transonic, and the well-known subsonic leading-edge vortex and vortex burst.

The specific results presented herein involve two 60° delta wing configurations (the configuration details can be found in the references from which the data are taken). The shuttle design has been changing so rapidly over the past year or two that is it impossible to present results for the "current" configuration. The aim of this paper is to point out general

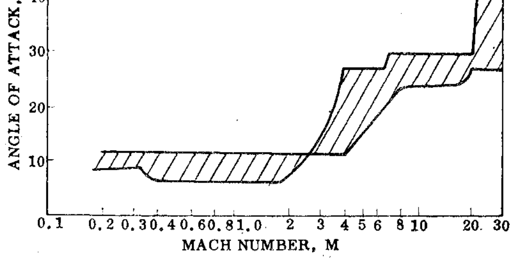


Fig. 1 Envelope of shuttle entry attitude histories.

areas of concern relative to the vehicle dynamic stability and to suggest possible means of eliminating or minimizing any adverse effects.

Discussion

The high angle-of-attack requirements probably cause the most severe aerodynamic stability problems inasmuch as the leeward wing flow is separated, or nearly separated, for much of the trajectory. Flying near incipient stall (either stalled or unstalled) is very dangerous because small perturbations in the flight attitude or yaw, or both, can cause a sudden, discontinuous change in stability that can raise havoc with the flight dynamics.

Leeside Shock-Induced Separation

Perhaps the most surprising unsteady flow phenomenon raced by the shuttle designer is the leeside shock-induced separation at hypersonic speeds. Contrary to the expectations from Newtonian theory, Seegmiller's excellent oil-flow photographs[1] show that at moderate angles of attack a significant region of attached flow exists on the leeward side of the delta wing. The leeside flow eventually separates from the wing, reattaching on the sides of the fuselage. Another zone of flow reattachment also occurs on the top of the fuselage owing to the flow entrained by the forebody vortices. The flow sketch in Fig. 2 illustrates the salient features of Seegmiller's flow visualization results. Notice that the vented wing separation region resembles the familiar subsonic leading-edge vortex although it results in a very different pressure distribution as will be discussed later.

Cross has classified the various hypersonic leeside flow phenomena.[2] His data suggest a number of distinct flows on the leeside of a delta wing at hypersonic speeds, three of which are applicable to the shuttle, i.e., types 1–3. At low angles of attack the flow converging from opposite wing panels (in the case of pure delta wing) is turned parallel to the freestream via a Mach wave (type 1). When the angle of attack is great enough to cause a detached leading-edge shock, the embedded leeside shock becomes strong enough to separate the boundary layer (type 2). That is, the subsonic flow aft of the detached leading-edge shock expands around the leading edge, reattaining supersonic speeds on the leeward side. The flow is still constrained to turn downstream near the wing root as before. However, this turning is now accomplished by a strong shock that causes the boundary layer to separate. As angle of attack is increased further, the leeside boundary layer is weakened. The increased terminal shock strength, due to increased leeside expansion, promotes separation. The separation region, therefore, grows until it reaches the leading edge (type 3). This shock-augmented, leading-edge separation is somewhat similar to the subsonic delta wing flow with a vortex bound to the leading edge. Increasing the angle of attack still further causes a breakdown of the bound vortex near the trailing edge. Finally, at still larger angles of attack, the leeward-side flow separation takes on a complicated three-dimensional, wakelike character.

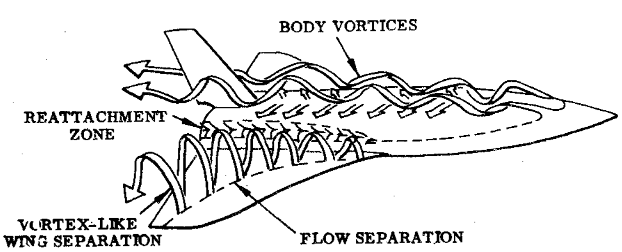


Fig. 2 Hypersonic leeside flowfield.

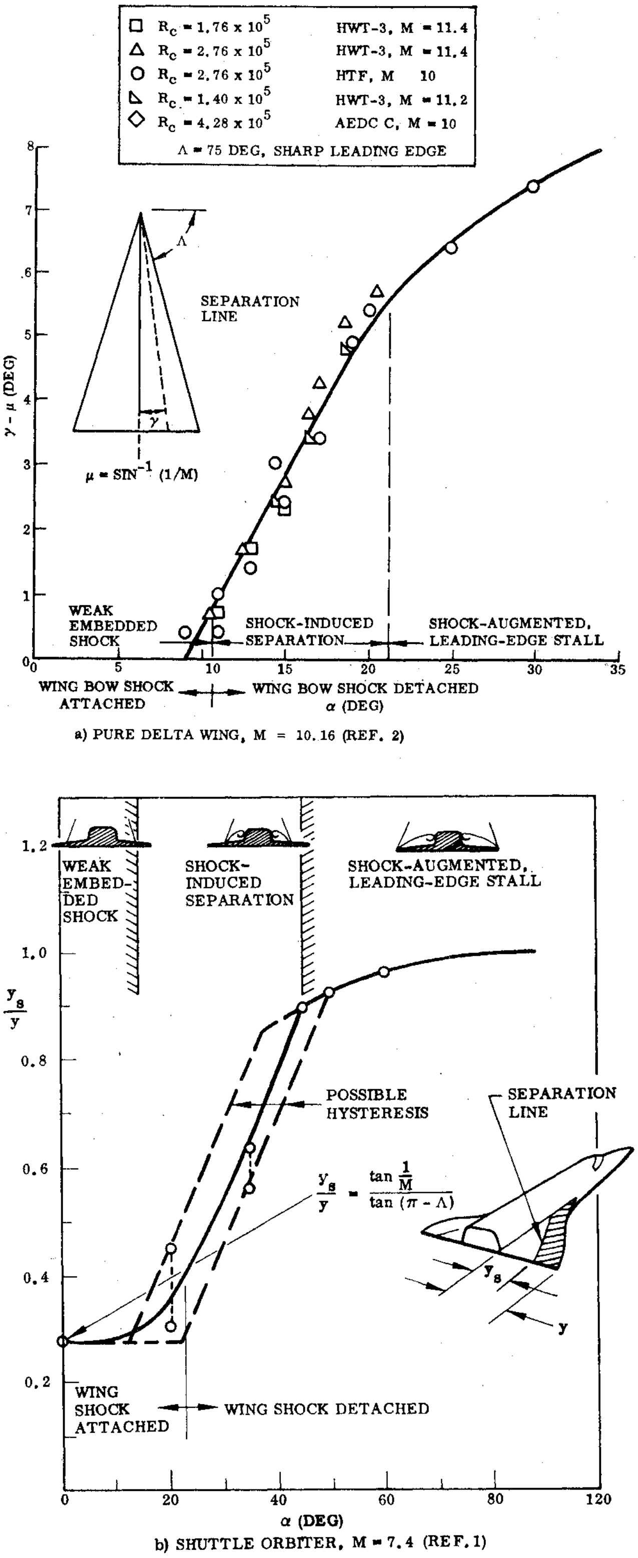


Fig. 3 Comparison of hypersonic leeside flow boundaries.

The changes in flow type observed by Cross correlate with discontinuities in the α-dependence of the terminal shock location. The shock position is determined by the Mach angle until leeside shock-induced separation occurs. The shock then moves outboard rapidly at constant rate until it has reached the leading edge (Fig. 3a). The delta wing space shuttle orbiter experiences similar flow phenomena (Fig. 3b). Initially, the flow is attached and the shock position corresponds to the Mach angle. When the angle of attack approaches the value for wing shock detachment,

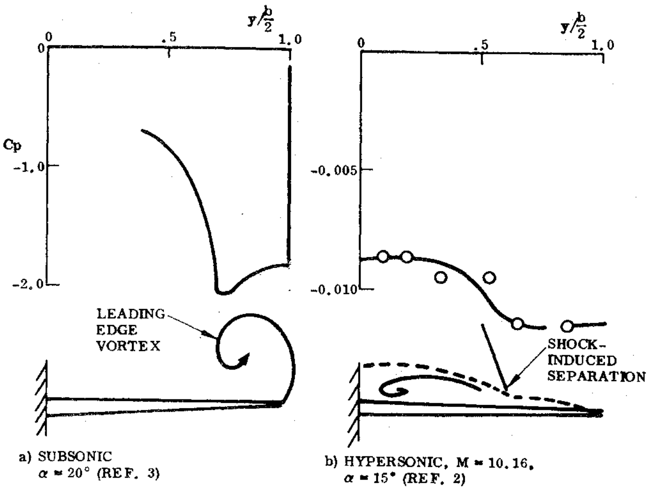


Fig. 4 Comparison of subsonic and hypersonic separation pressures.

the separation shock moves rapidly outward. The scatter in the shock position measurements may be indicative of flow hysteresis.

Although the flow in the separated region resembles the subsonic leading-edge vortex, the pressure distribution does not. There is a pressure rise associated with the occurrence of hypersonic shock-induced separation as opposed to the large negative pressure associated with a subsonic leading-edge vortex (Fig. 4).[2,3] Thus, the hypersonic leeside shock-induced separation tends to decrease the wing lift, where as the subsonic leading-edge vortex increases the lift.

As one would expect, the changes in the leeward-side flow patterns affect the stability derivatives. The roll stability is affected most. Figure 5 shows that a rapid increase in the roll stability of the wing occurs in the region of shock-induced separation. These results represent the wing roll stability in the presence of the fuselage. Note that the increase in stability begins before the theoretically determined angle of attack of wing bow shock detachment. Figure 3b indicates also that the shock begins its outboard movement before the theoretical detachment attitude is reached. This is primarily the result of fuselage interference. The fuselage width is increasing along the wing root chord for this configuration.[4] Thus, the flow at the root is constrained to turn outboard parallel to the fuselage rather than downstream. This causes a relatively stronger shock and an earlier separation. The wing bow shock also tends to detach earlier than predicted because the predictions assume a sharp leading edge and that of the shuttle wing is blunt. Furthermore, it seems likely that a corner vortex could occur in the thick viscous layer at the wing body juncture, which would also tend to cause early separation.

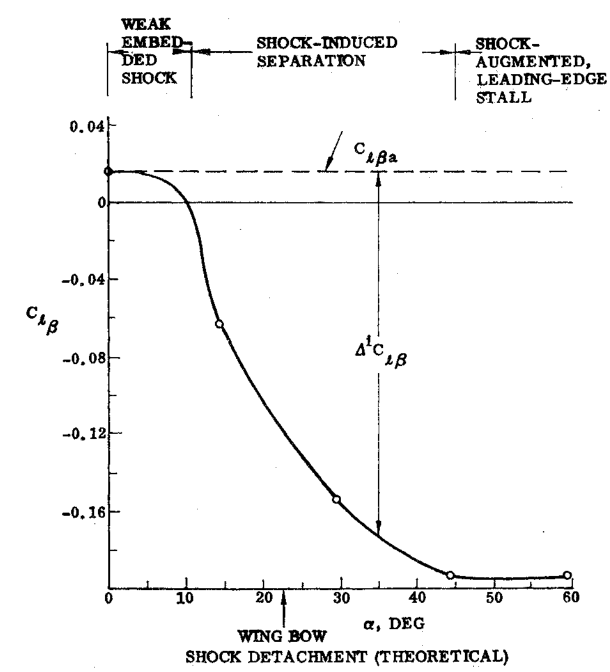


Fig. 5 Local and induced roll derivatives on a shuttle wing, $M = 7.4$ (Ref. 4).

The fuselage causes a considerable increase in the pressures in the region of shock-induced separation. By assuming that the fuselage acts as a step to the incoming flow component normal to the leading edge, a "strip" pressure coefficient of 0.29 can be realized at $M = 7.0$.[5] This corresponds to $\Delta Cp_s = 0.0725$, based on freestream conditions, which is an order of magnitude greater than the plateau pressure rise on a pure delta wing (Fig. 4b).[2] Thus, the leeside effects are significant.

Yawing the shuttle effectively reduces the angle of attack on the windward wing, causing a contraction of the separation. A corresponding increase of the separation results on the leeward wing. Thus, the windward wing experiences a lift increase, and a corresponding lift reduction occurs on the leeward wing. The result is the observed large stable $C_{l\beta}$. This change in wing loading is not realized instantaneously, but involves a flowfield timelag, as is usually the case when flow separation is involved. Thus, the stable rolling moment due to the leeside shock movement will lag the vehicle motion dynamically.

Consider the shuttle flying in the angle-of-attack range of shock-induced separation and simultaneously describing harmonic oscillations in roll about $\phi = 0$ (Fig. 6). Ordinarily as the vehicle rolls through $\phi = 0$, the flow patterns will be symmetric (dashed shock position in Fig. 6a) and the rolling moment zero. However, as a result of the flowfield timelag, the instantaneous separation configuration at $\phi(t) = 0$ will be that of an earlier instant when the vehicle had the roll attitude $\phi(t - \Delta t)$ (Fig. 6b). Thus, dynamically, a residual asymmetric wing loading occurs at $\phi = 0$, which results in a residual rolling moment in the direction of the motion; that is, it is undamping (Fig. 6a). Thus, the hypersonic leeside shock-induced separation will reduce the roll damping,

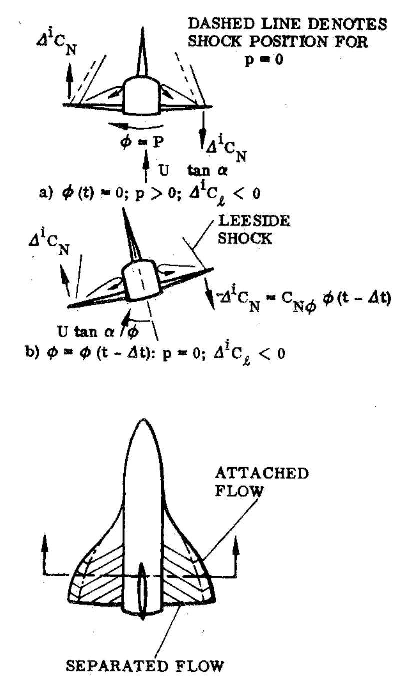


Fig. 6 Effect of shock-induced separation on roll damping.

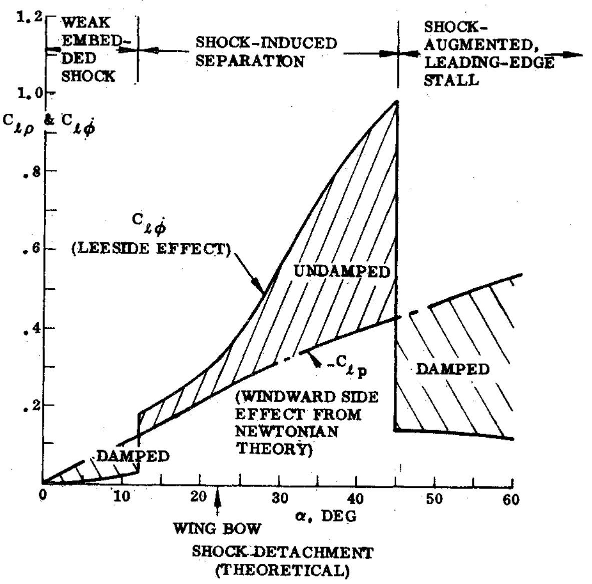


Fig. 7 Estimated roll damping at $M = 7.4$.

and perhaps even cause undamping as a result of the flowfield timelag. A similar phenomenon happens at subsonic speeds when the lift on the delta wing is dominated by the leading-edge vortex. The finite convection speed of the leading edge vortex supplies the timelag, which results in a measured damping that is quite different from attached flow predictions. However, the sign of the vortex-induced normal load is positive (opposite that of hypersonic shock-induced separation, Fig. 4), which results in a more stable damping derivative (also opposite to the hypersonic results).[6]

An estimate has been made of the effect of the hypersonic leeside shock-induced separation on the shuttle roll damping. This is compared with the windward-side damping effect estimated from Newtonian theory (Fig. 7)‡ The results show that undamping can occur as the result of shock-induced separation. The computations essentially followed the technique used so successfully in estimating the unsteady aerodynamics of the Apollo-Saturn booster.[10] This work showed that the dynamic effect of the separated flow load is proportional to the product of the static moment and the timelag; i.e., for the present case $\Delta' C_{l\dot\phi} = -C_{l\phi}\Delta t\,(2U/b)$. The static moment was obtained from Cleary's results (Fig. 5),[4] making use of the relation $\partial C_l/\partial\phi = (\partial C_l/\partial\beta)\sin\alpha$ for small ϕ and β. The timelag differs in the different flow regimes. In the critical shock-induced separation region the flow was assumed to be similar to a laminar shock-augmented stall (the leeside boundary layer is laminar at separation over most of the hypersonic portion of the trajectory). That is, the flowfield normal to the leading edge resembles that of an airfoil in laminar, transonic flow. When the bow shock detaches, the leading-edge flow is subsonic. The flow reaches supersonic speeds locally as it expands around to the leeward side of the wing. There it is again decelerated to subsonic speeds by the strong separation shock. In this case the timelag has three components: $\Delta t = \Delta t_s + \Delta t_M + \Delta t_w$. Δt_s is the laminar separation point lag, and Δt_M is the apparent Mach number effect on shock strength due to the shock oscillation velocity.[7] The last term, Δt_w, is the usual von Kármán-Sears wake circulation lag for a simple-two-dimensional airfoil. This is analogous to the time required for the vortexlike flow aft of the separation shock to respond to an attitude perturbation. Since the flow resembles a leading-edge vortex physically, it was assumed that the timelag was also similar; i.e., $\Delta t_w = c/U$.[11]

Outside the critical shock-induced separation region the shock position was assumed to be a function of the local corner flow details (similar to step-induced separation) for the case of a weak embedded shock; or locked to the leading edge (similar to nose-induced separation) for shock-augmented leading-edge stall.[7] In both cases the separation responds to perturbations in the thickness and velocity profile of the approaching boundary layer and to backpressure changes in the recirculation region.[10] The former lags the vehicle motion owing to the finite boundary-layer convection speed: $\Delta t_{b.l} = l_\perp/0.8\,U_\perp$, and the latter owing to the vortex convention speed $\Delta t_w = c/U$. Thus, $\Delta t = \Delta t_{b.l} + \Delta t_w$.

The roll damping predictions, in spite of their approximate nature, are sufficiently representative to indicate that futher analytic and experimental efforts should be directed toward obtaining a better evaluation of the possibility of roll undamping at hypersonic speeds. Estimates of the effects of shock-induced separation on pitch and yaw damping showed no similar potential for dynamic instability.

‡ Newtonian theory gives a better estimate of the windward-side effect for $\alpha \gg 0$ than the hypersonic small disturbance theory used before.[8,9] Also, the values for the leeward-side effect in Fig. 7 are about double those shown previously[8,9] since $b/2$ is now used as the reference length.

Sudden Leading-Edge Stall

When the Mach number normal to the leading edge is slightly less than unity ($M_\infty = 2.0$ for the 60° swept shuttle wing), the separation can suddenly switch from the shock-induced variety to leading-edge separation, with a corresponding discontinuous change in wing loading. This phenomenon is analogous to the switch between transonic flow attachment and leading-edge stall and is, therefore, dependent upon the airfoil section configuration.[12] Typical boundaries for sudden leading-edge stall for a practical airfoil section are shown in Fig. 8. The disconcerting feature of this plot is the jump from an aft shock position (transonic attachment) to leading-edge separation, implying a very large change in loading.

For some rather thin airfoils the flow was observed to oscillate between leading-edge separation and transonic attachment in a narrow α-M region. This behaviour is reminiscent of the results of Chevalier and Robertson.[13] They observed that the flow aft of the shoulder of a cone-cylinder body alternated between totally separated and attached. The body of revolution results are simply the three-dimensional analog of the airfoil results. The discontinuous jump from attached to separated flow occurs when the terminal normal shock enters the near nose region with its adverse pressure gradient. At some point the shock-induced pressure rise, coupled with the near nose adverse pressure gradient, causes the separation to jump to a point where the boundary layer encounters a pressure gradient that it can tolerate (to the shoulder in the case of the cone-cylinder).

The similarity between the two- and three-dimensional flows is illustrated in Fig. 9. In both cases, large, discontinuous, statically stabilizing, pitching moments result when the sudden nose stall is established. For the cone-cylinder body it has been shown that, dynamically, the jump will lag

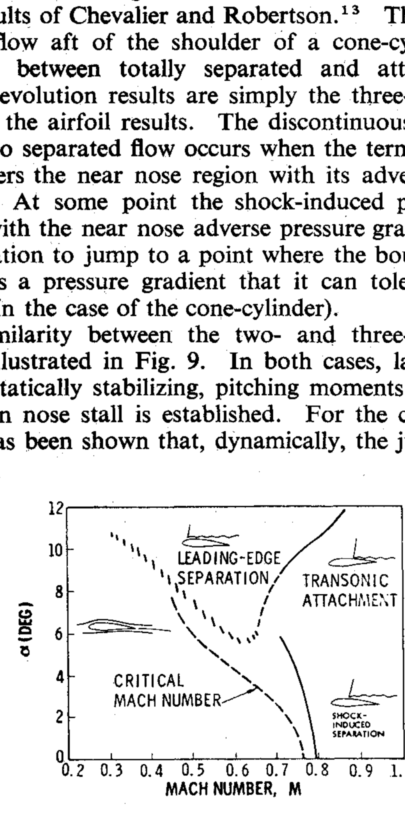
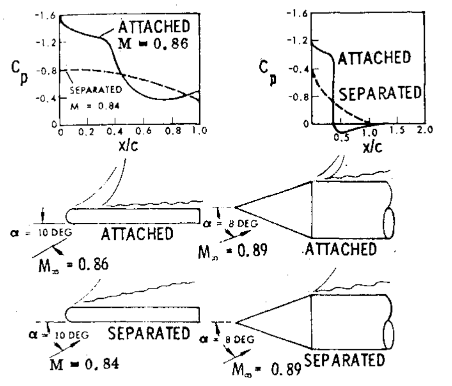


Fig. 8 Sudden leading-edge stall (Ref. 12).



a) SLAB WING (REF. 12) b) 15-DEG CONE CYLINDER (REF. 13)

Fig. 9 Comparison of pressure distributions for sudden stall.

the body motion, owing mainly to the accelerated flow-relief of the adverse pressure gradient and, to a lesser extent, owing to the delay of boundary-layer buildup on the leeward side (as a result of a finite convection speed in the boundary layer).[14,15] Likewise, pitch rate-induced camber and accelerated flow effects will delay the jump to leading-edge stall.[16] The jumpwise change in the aerodynamic load produces a large increase of the stable static moment derivative. However, because of the timelag, the jump generates a large undamping derivative for small amplitude oscillations at the jump angle of attack. As the oscillation amplitude is increased, the finite stability derivatives on either side of the jump begin to moderate the effect of the jump and at some amplitude they balance the jump effect. The result is a limit cycle oscillation. It has been shown[15] that the aerodynamic undamping of the second elastic mode of the Saturn I with a Jupiter nose shroud[17] was the likely result of such sudden leeside separation. Sudden leeside separations on the shuttle will have equally drastic effects. That is, limit cycle oscillations in pitch or roll, or both, could result unless damping is supplied via the control system.

Leading-Edge Vortex and Vortex Burst

It is well-known that a major portion of the lift on delta wings at subsonic speeds is the result of the so-called leading-edge vortex.[18,19] At very high angles of attack the vortex becomes unstable, breaks down, or "bursts." The result is a loss of core suction which destroys the vortex lift and generates a violent pitchup as burst progresses forward from the trailing edge (Fig. 10).[20] The vortex breakdown is very sensitive to angle of attack, moving almost jump-wise from the trailing edge to the ⅔ chord, at least for high sweep angles (Fig. 11).[21] Likewise, burst location can be unstable, causing extreme buffet loads. It may even exhibit hysteresis effects.[22] Burst is very sensitive to sweep. Yaw effectively changes the sweep angle, decreasing it on the windward wing and increasing it on the leeward. This explains why burst occurs on the windward wing of the Concorde at greatly reduced angle of attack when the aircraft is yawed (Fig. 10).[20] Because of the violent nature of the effect of vortex burst on the shuttle stability characteristics, burst should be avoided. Fortunately, judging from a representative trajectory (Fig. 1), it appears that the shuttle will fly well below vortex burst (at least for $\beta = 0$). However, the vehicle can still experience burst under combined pitch and yaw conditions.

Control-Induced Effects

Heretofore the effect of back pressure on the leeside flowfield of the delta wing has not been considered. It is well-known that the extent of shock-induced separation is

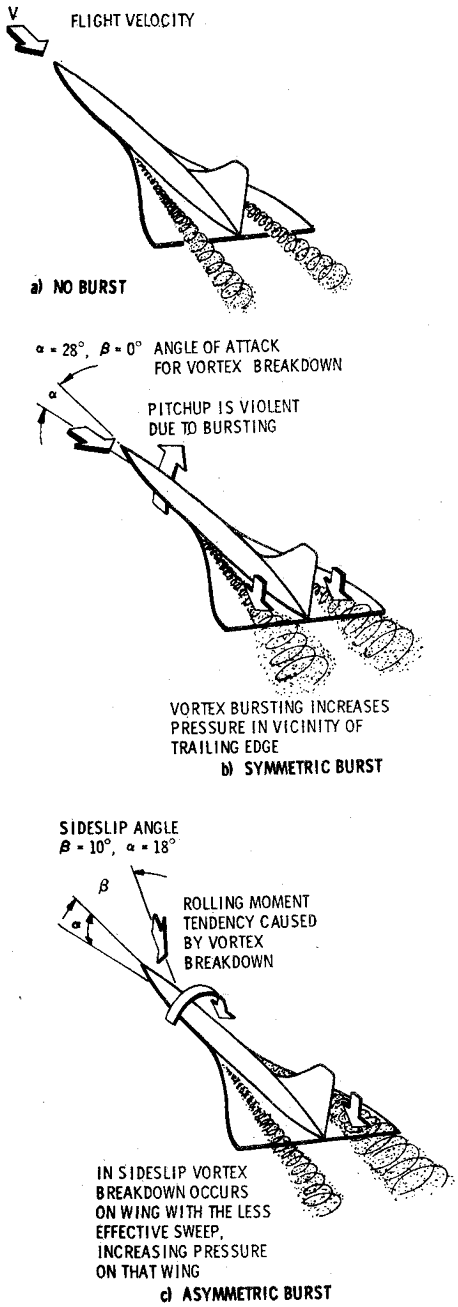


Fig. 10 Effect of vortex burst on stability (Ref. 20).

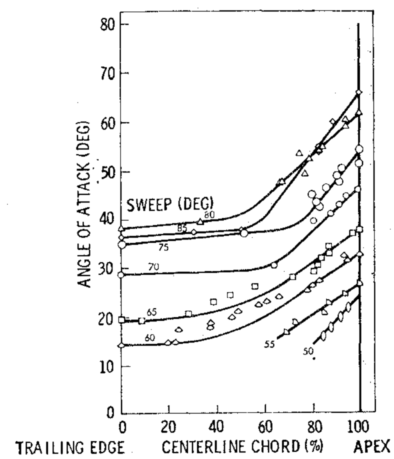


Fig. 11 Effect of angle of attack and sweep on burst (Ref. 21).

sensitive to back pressure.[23] Flap controls will often cause boundary-layer separation, especially in hypersonic low-density flow, where less than 10° flap deflection often is enough.[24,25] Thus, the deflection of a trailing-edge control surface will affect the extent of shock-induced separation. Such backpressure effects are of practical concern since it is desirable to control the shuttle with leeward control deflections, wherever possible, in order to minimize control curface heating. Data obtained on a typical orbiter show an elevon effectiveness greater than Newtonian for small deflections ($\delta = -10°$) at low angles of attack (Fig. 12).[26] This is the likely result of shock-induced separation of the leeside flow. The separation extent (for fixed-flap deflection $\delta = -10°$) increases initially with angle of attack. However, the back pressure effect from the flap causes the transition between shock-induced separation and leading-edge stall to start near the flap and progress forward with increasing angle of attack (see sketch in Fig. 12). At high angles of attack the flow separates from the leading edge, forming a leading-edge vortex. This results in a reversal of the surface flow direction over the flap as the wing is submerged in recirculating flow. Thus, the shock-induced separation forward of the flap vanishes. In addition, the flap is subjected to a minor suction pressure under the leading-edge vortex.[27] The result is a loss of flap effectiveness below the Newtonian value (i.e., a more stable ΔC_M than predicted by Newtonian theory for $\delta = -10°$). At the high deflection $\delta = -30°$ the mixed flowfield may still occur, but the over-all force data is not sensitive enough to detect it. Generally, the leeside effects seem to vary less drastically, and Newtonian theory seems to predict the α-trends rather well.

Experimental results also indicate that roll reversal occurs as a result of the backpressure-induced flowfield change.[28] If hysteresis does occur when switching between the various separated flow types, then a residual control will remain after the control deflection is removed. The control force is made up of two components: the force on the control surface itself and the induced load on the wing owing to a control-induced change of the flowfield (separation type) on the wing. The former will go away when the control deflection is removed; the latter will persist (if flowfield hysteresis is present) until the angle of attack is reduced sufficiently to get out of the hysteresis region.

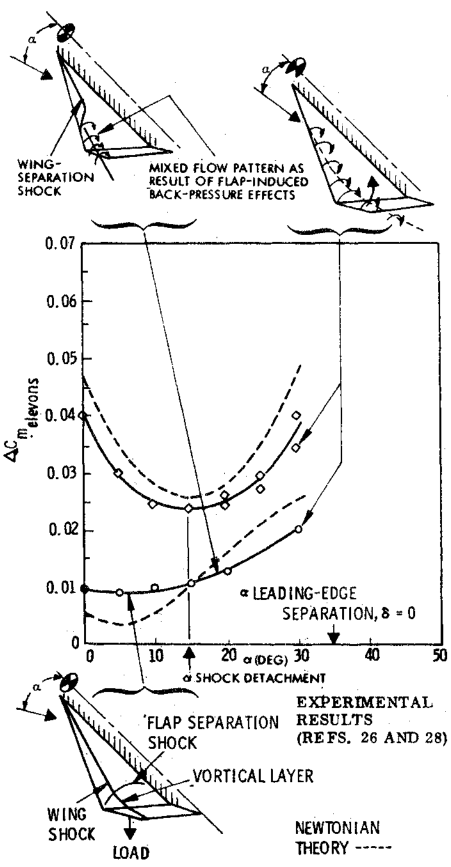


Fig. 12 Hypersonic control inteference ($M = 5.0$).

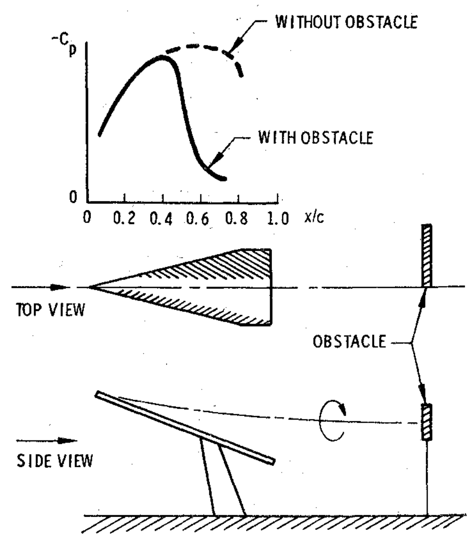


Fig. 13 Vortex burst induced by a downstream obstacle (Ref. 30).

As one would expect, vortex burst is also sensitive to backpressure. In agreement with Ludwieg's theory[29] Hummel finds that supplying an adverse pressure gradient by using an obstacle one chord length downstream of the trailing edge on the right half span causes vortex breakdown (Fig. 13).[30] An upward flap deflection, e.g., for a roll maneuver, will, of course, have a similar effect, thereby causing a "super response" to a roll command. Thus, control deflection will induce burst, where ordinarily it would not occur. Furthermore, vortex burst is definitely associated with hysteresis, and subsonic-control-induced burst is a problem of concern.

The shuttle configuration differs from previous delta wing aircraft in that the fuselage is considerably larger relative to the wing. This, coupled with the large leeward control deflections, results in some unusual interference effects. Preliminary results from recent tests§ on the MSC 040A configuration reveal a discontinuity in the longitudinal and lateral moments at $\beta = 0$ (Fig. 14). The yaw discontinuities (in moment characteristics) observed at $M = 0.9$ occur at the subsonic cruise angle of attack (Fig. 1). The effect was present to some degree for $0.6 \leq M \leq 1.46$ at $\alpha = 10°$. Thus, it occurs over a large portion of the trajectory and could have significant effect on the shuttle dynamics. Double discontinuities could occur if both elevons were deflected in a combined pitch-roll command. The violent nature of these effects label them as something to be avoided, if possible, just as was vortex burst. The discontinuous, unstable, roll effect raises the possibility of snap roll similar to that which plagued the straight wing shuttle.[31] The phenomenon warrants further analytical and experimental study.

Avoiding the Problems

Even though the effects of separation on shuttle dynamics may be formidable, the elimination of these problems can be carried out in a straightforward manner. First, the various unstable separated flow boundaries need to be mapped for

§ These tests were specifically designed to explore the effect of control deflection on the leeside wing flowfield. The results will be analyzed and reported in a study conducted for the NASA Marshall Space Flight Center, Contract NAS-8-28130, under the direction of W. W. Clever.

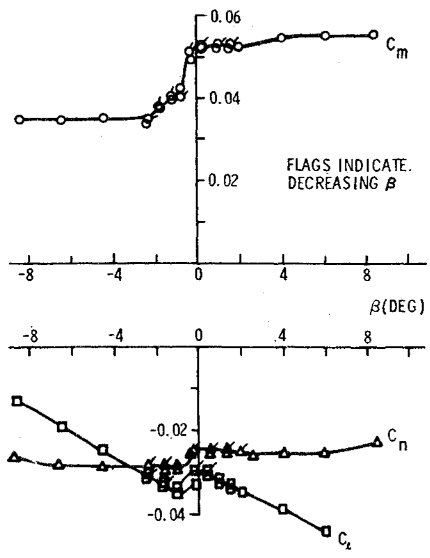


Fig. 14 Discontinuous yaw characteristics ($M = 0.9$, $\alpha = 10°$).

realistic combinations of M, α, β, and δ. A word of caution relative to the definition of the unsteady flow boundaries is appropriate at this time. Wind-tunnel results alone may not give a true indication of the unsteady flow boundaries. There is the usual difficulty in assuring proper Reynolds number simulation, which is even more difficult when flow separation is involved. There is also the more unusual problem of simulating the heat addition to the boundary layer from the hot shuttle structure. The hot structure is, of course, the result of earlier heating during entry. Recent evidence shows that the hot wall can promote separation.[32] Thus, defining realistic unsteady flow boundaries is not an easy task.

After the flow boundaries are defined, the next task is to explore these regions to ascertain just how serious these effects really are. Once the regions of severe instability are defined, the trajectory can be shaped to minimize the realization of adverse effects. Two such trajectories are shown in Fig. 15. These trajectores are ficticious, constructed only to illustrate the philosophy of minimizing adverse separation effects. They do not necessarily comply with crossrange or heating constraints. Generally, the approach is to fly well away from the critical flow boundaries to avoid the involuntary realization of unstable flow due to excessive control deflection, gust, etc. When it is necessary to enter an unstable flow region, it should be passed through as quickly as possible. Furthermore, it is prudent to fix separation on both wings

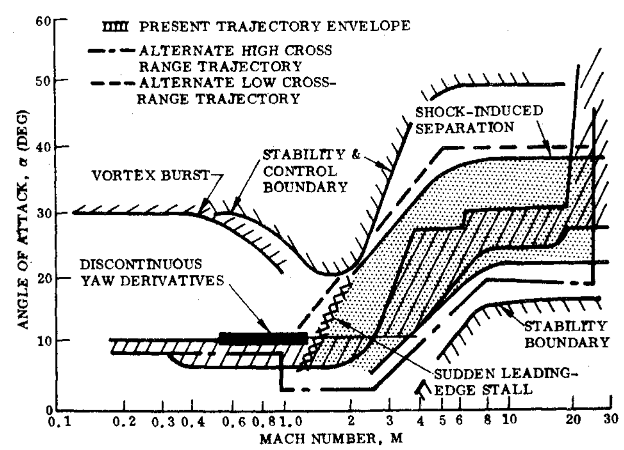


Fig. 15 Minimizing unsteady aerodynamic problems with trajectory shaping.

simultaneously before traversing a separated flow region, and afterwards cause the flow to attach simultaneously on both wings, thus avoiding the possibility of snap roll as was experienced on the straight wing orbiter.[31] This can be accomplished by deploying properly designed spoilers prior to entering an unstable separated flow region and retracting them when well out of the region. This is particularly desirable when traversing regions of sudden stall and vortex burst. In order to prevent delta wing flow separation effects from dominating the shuttle dynamics, one must first recognize that this a real possiblity and then set about to eliminate theses unsteady flow problems as the design evolves. This can eliminate possible nasty surprises, particularly for the pilot, during the flight test program.

As the shuttle re-entry trajectories are rather tightly banded by instability regions, it seems prudent to try to gain more leeway in order to allow for gusts, sudden maneuvers, pilot error, etc. Planform modifications have long been recognized as the most effective way of altering delta wing stall boundaries. Decreasing the sweep angle delays leeside shock-induced separation. The reason is obvious. Reducing sweep increases the Mach number normal to the leading edge, thus delaying bow shock detachment and associated shock-induced separation. Reduced sweep, however, brings the subsonic vortex burst down to lower angles of attack. Judging from Fig. 15, a moderate reduction might be acceptable. In order to allow more leeway for yaw, control deflection, and landing flare, a double-delta planform might be in order. If correctly designed, it could augment stability and delay burst [21] and in addition, it could fix forebody vortex shedding and, thereby, eliminate other destabilizing effects observed on the shuttle.[8,9]

Conclusions

A study of the unsteady aerodynamics of the high crossrange shuttle orbiter indicates that the shuttle will be subjected to three delta wing separation phenomena that could dominate the vehicle dynamics. They are 1) leeside shock-induced separation, 2) sudden leading-edge stall, and 3) subsonic leading-edge vortex and vortex burst. These effects are due, primarily, to the high angles of attack required of the shuttle. They are considerably complicated by control deflections.

The time spent in the unstable flow regions should be minimized. That is, the penetration of these regions should be done as quickly as possible by proper tailoring of the trajectory, with separation fixed to avoid asymmetric separation or attachment that can produce snap roll. A great deal can be done with planform oprimization and minor geometry changes to avoid or minimize the problems.

A review of the unsteady flow problems of the delta wing shuttle orbiter indicates that there is every reason to believe that the separation-induced problems can be dealt with successfully. However, this will require a careful analysis of the vehicle motion, including the effects of the strong pitch-yaw-roll coupling.

References

[1] Seegmiller, H. L., "Surface Flow Visulization Investigation of a High Cross Range Shuttle Configuration at a Mach Number of 7.4 and Several Reynolds Numbers," TMX-62,036, June 2, 1970, NASA.

[2] Cross, E. J., "Analytical Investigation of the Expansion Flow Field over a Delta Wing at Hypersonic Speeds," ARL 68-0027, Feb. 1968, Aerospace Research Lab., Wright-Patterson Air Force Base, Ohio.

[3] Wentz, W. H., Jr. and McMahon, M. C., "Further Experimental Investigations of Delta and Double-Delta Wing Flow Fields at Low Speeds," CR-714, Feb. 1967, NASA.

[4] Clearly, J. W., "Hypersonic Stability and Control Characteristics of a High-Cross Range Orbital Vehicle," TMX-62,065, May 22, 1970, NASA.

[5] Speaker, W. V. and Ailman, C. M., "Static and Fluctuating Pressures in Regions of Separated Flow," AIAA Paper 66–456, Los Angeles, Calif., 1966.

[6] Boyden, R. P., "Effects of Leading-Edge Vortex Flow on the Roll Damping of Slender Wings," *Journal of Aircraft*, Vol. 8, No. 7, July 1971, pp. 543–547.

[7] Ericsson, L. E. and Reding, J. P., "Unsteady Airfoil Stall and Stall Flutter," CR-111906, June 1971, NASA.

[8] Reding, J. P. and Ericsson, L. E., "Review of Delta Wing Space Shuttle Dynamics," *Space Shuttle Aero Thermodynamics Techonology Conference, Volume III-Aerodynamics*, TMX-2508, Feb. 1972, NASA, pp. 861–921.

[9] Reding, J. P. and Ericsson, L. E., "Review of Delta Wing Space Shuttle Vehicle Dynamics," Rept. LMSC-D2432938, Oct. 1971, Lockheed Missiles & Space Company Inc., Sunnyvale, Calif.

[10] Ericsson, L. E. and Reding, J. P., "Analysis of Flow Separation Effects on the Dynamics of a Large Space Booster," *Journal of Spacecraft and Rockets*, Vol. 2, No. 4, July–Aug. 1965, pp. 481–490.

[11] Lamborne, N. C., Bryer, D. W., and Mayberg, J. F. M., "A Preliminary Note on the Behavior of the Leading-Edge Vortices of a Delta Wing Following a Sudden Change of Incidence," AERO Note 1006, March 1962, National Physics Lab., Teddington, Great Britain.

[12] Lindsey, W. F. and Landrum, E. J., "Compilation of Information on the Transonic Attachment of Flows at the Leading Edges of Airfoils," TN 4204, Feb. 1958, NACA.

[13] Chevalier, H. L. and Robertson, J. E., "Pressure Fluctuations Resulting from Alternating Flow Separation and Attachment at Transonic Speeds," TDR 63–204, Nov. 1963, Arnold Engineering Development Center, Tullahoma, Tenn.

[14] Ericsson, L. E., French, N. J., and Guenther, R. A., "The Aeroelastic Characteristics of the Saturn 1B Launch Vehicle with Biconic Payload Shroud," Rept. M-37-67-1, NASA Contract NAS 8-11238, July 1967, Lockheed Missiles & Space Company Inc., Sunnyvale, Calif.

[15] Ericsson, L. E., "Aeroelastic Instability Caused by Slender Payloads," *Journal of Spacecraft and Rockets*, Vol. 4, No. 1, Jan. 1967, pp. 65–73.

[16] Ericsson, L. E. and Reding, J. P., "Unsteady Airfoil Stall, Review and Extension," *Journal of Aircraft*, Vol. 8, No. 8, Aug. 1971, pp. 609–616.

[17] Hanson, P. W. and Dogget, R. V., Jr., "Aerodynamic Damping and Buffet Response of an Aeroelastic Model of the Saturn I Block II Launch Vehicle," TND-2713, March 1965, NASA.

[18] Rogers, W. E., Berry, C. J., and Townsend, J. E. C., "A Study of the Effects of Leading Edge Modifications on the Flow over a 50-Degree Sweptback Wing at Transonic Speeds," R & M 3270, 1962, Aeronautical Research Council, Great Britain.

[19] Polhamus, E. C., "Application of the Leading-Edge Suction Analogy of Vortex Lift to the Drag Due to Lift of Sharp-Edge Delta Wings," TND-4739, Aug. 1968, NASA.

[20] Sforza, P. M., "Aircraft Vortices Benign or Baleful?," *Space-Aeronautics*, April 1970, pp. 42–48.

[21] Wendtz, W. H. and Kohleman, D. L., "Vortex Breakdown on Slender Sharp-Edged Wings," AIAA Paper 69–778, Los Angeles, Calif., 1969.

[22] Lowson, M. V., "Some Experiments with Vortex Breakdown," *Journal of the Royal Aeronautical Society*, Vol. 68, May 1964, pp. 343–346.

[23] Needham, D. A. and Stollery, J. L., "Boundary-Layer Separation in Hypersonic Flow," AIAA Paper 66-455, Los Angeles, Calif., 1966.

[24] Keating, R. F. A. and Mayne, B. L., "Low-Speed Characteristics of Waverider Wings," TR 69051, 1969, Royal Aircraft Establishment, Great Britain.

[25] Kirkpatrick, D. L. I. and Field, J. D., "Experimental Investigation of the Positions of the Leading Edge Vortices Above Slender Delta Wings with Various Rhombic Cross-Sections in Subsonic Conical Flow," TR 66068, 1966, Royal Aircraft Establishment, Great Britain.

[26] Freeman, D. C., "Supersonic Aerodynamic Stability, Control, and Performance of a Modified NR-134D Orbiter Configuration," SADSAC DMS-DR-1101, June 1971, NASA.

[27] Rao, D. M. and Whitehead, A. H., Jr., "Lee-Side Vortices on Delta Wings at Hypersonic Speeds," *AIAA Journal*, Vol. 10, No. 11, Nov. 1972, pp. 1458–1465.

[28] Goldberg, G., Emery, C., Olsen, R., and Munnell, R., "Hypersonic Stability and Control Investigation and Evaluation of Split Elevon Concept of Yaw Control for the 0.00763 Scale NR Delta Wing Orbiter, NAR 134D/161B," SADSAC DMS-DR-1095, May 1971, NASA.

[29] Ludwieg, H., "Zur Erklärung der Instabilität der uber angestellten Deltaflügeln auftretenden freien Wirbelkerne," *Zeitschift fur Flugwissenschaften*, Vol. 10, 1962, pp. 242–249.

[30] Hummel, D., "Untersuchungen über das Aufplatzen der Wirbel an schlanken Deltaflügeln," *Zeitschrift fur Flugwissenshaften*, Vol. 13, 1956, pp. 158–168.

[31] Hamilton, E. J. and Ryals, W. G., "Review of 1/10 Scale Space Shuttle Air Drop Test Results," MSC Internal Document 5-2950-1-NOU-126, June 1970, NASA.

[32] Macha, J. M., Norton, D. J., and Young, J. C., "Surface Temperature Effect on Subsonic Stall," AIAA Paper 72–960, Palo Alto, Calif., 1972.