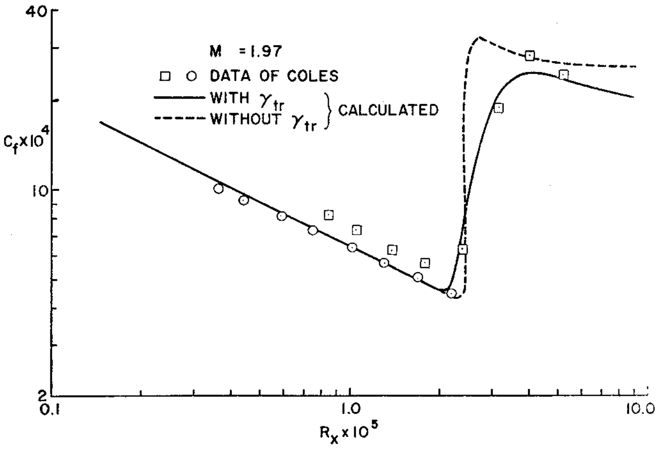

Fig. 4 Comparison of calculated and experimental skin-friction coefficients for a supersonic flat-plate flow measured by Coles.

of Schubauer.[13] This is an incompressible flow at relatively low Reynolds numbers. Figure 4 shows the results for the experimental data of Coles.[14] This is a supersonic adiabatic flow. The results indicate that in both cases the calculations by using the intermittency distribution given by Eq. (10) seem to account for the transition region rather well.

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Alleviation of Vortex-Induced Heating to the Lee Side of Slender Wings in Hypersonic Flow

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Recent hypersonic studies show that the vortex system over the lee surface of planar delta wings,[1–3] blunted half-cones,[4,5] and on a conceptual high-cross range shuttle vehicle[6] can cause intense heating to the centerline region. The critical effect of Reynolds number in determining the location, initiation, and intensity of the vortex-induced heating has been studied.[4–6] Moore[7] indicates that vortices are produced at a leading edge corner, i.e., wherever a discontinuity exists in the leading-edge geometry. This observation led the present authors to propose contouring the planform of the apex region of a slender delta wing to more gradually turn the flow to reduce the interaction between opposing leading edge flows, which leads to the formation of the vortex system.

The present study was done in the Langley 11-in. Mach 6.8 Tunnel employing a sharp-apex delta wing, a rounded (circular-arc) apex delta wing, and hyperbolic and parabolic-planform wings. The leading edges of the models were sharp (<0.075 mm thick) with flat upper surfaces and lower surface bevel angles (perpendicular to the leading edge) between 18° and 20°. The sharp and rounded-apex delta wing models used in the oil-flow study were swept 75° so these results together with those reported in Refs. 1–3 provide information over at least a small range of sweep (Λ = 70°, 75°, 78°) for comparison with the hyperbolic and parbolic-planform wing results. The delta wings employed in the heating and vapor screen tests were swept 70°. Some details of the models are shown in Figs. 1 and 3. For the delta wings, the length L is the total length and for the hyperbolic and parabolic shape wings, L represents the distance along the root chord to the

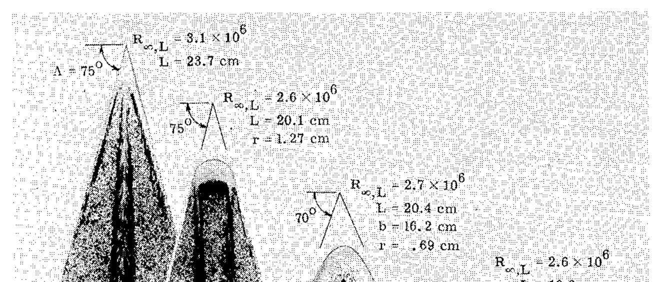

Fig. 1 Oil flow and Schlieren photographs; α = 7°; M = 6.8.

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point of maximum span b. The radius of curvature of the models at the apex is denoted by r. The set of models used for heat-transfer measurement and vapor screen tests (except the parabola) had a section removed from the steel planar surface, and the cavity filled with insulating material suitable for heating tests using the phase-change paint technique.[8] The oil flow and phase-change paint tests were run at a stagnation temperature of 620°K; the vapor-screen tests were run with the supply air at room temperature.

The shock detaches from the leading edge at the apex of the 70° and 75° sharp-apex delta wings, just upstream of the tangency point of the rounded-apex delta wings, and ahead of the 1-in. station on the hyperbolic planform wings for the angles of attack under consideration; the detachment point (Λ_{SD}) on the parabolic wing at an angle of attack (α) of 7° is shown in Fig. 1. The flow-visualization techniques employed (oil flow and vapor screen) are discussed in more detail in Ref. 2. At $\alpha = 7°$, the flow over the lee side of the 75° delta wing separates at the leading edge, except in the apex region, whereas the flow separates inboard of the leading edge on the 70° swept wing (see Fig. 9 of Ref. 2). However, a similar influence of the vortex system on the centerline heating and shear has been found on both delta wings[1–3]; the featherlike oil flow trace shown in Fig. 1 typifies the results.

Since the vortex initiates near the apex of the delta wing, a first attempt to influence the production of the vortices was to round the apex planform shape with a circular arc while keeping the leading edge sharp. The oil flow results in Fig. 1 show the inadequacy of this approach. Vortices form at the tangency point with a strong downstream influence. This curvature discontinuity was removed by providing a continuous planform curvature generated by parabolic or hyperbolic contours. The oil flow shown for these configurations when compared to the delta wing results indicate a large shear reduction in the center region, though an increase in shear occurs in the leading-edge region. The location of shock detachment on the parabola, shown by the schlieren in Fig. 1, was thought to be a possible source of vortex initiation. However, the oil flow does not indicate any vortices downstream of this detachment.

Heating patterns obtained on the insert of these wings at $\alpha = 10°$ corroborate evidence provided by the oil flow patterns. The centerline heating (freestream Stanton number, St_∞) on the sharp-apex delta, shown in Fig. 2, is some five times the value predicted by laminar theory. Unpublished thermocouple data shown in the figure was obtained by the senior author on a comparable 70° swept delta in the Langley 20-in. Mach 6 Tunnel (stagnation temperature = 530°K), and indicates the same heating level and distribution as that obtained by the phase-change paint technique. The observed distribution with freestream Reynolds number ($R_{\infty,x}$) seems characteristic of transition, yet caution must be observed in attributing boundary-layer characteristics to such a vortex-dominated flowfield. For example, heating data obtained on a 75° sweep[1] and a 70° sweep[2] delta wing at $\alpha = 5°$ showed two peaks down the leeward centerline, the higher peak occurring close to the origin of the vortex system and the second peak attributed to boundary-layer transition. A more fundamental understanding is needed of the effect of transition and turbulent flow on the vortex-induced effects and, in turn, the effect of the vortex presence on the transition process. Because of the limited data-gathering area on the present models, heating data is lacking near the apex region.

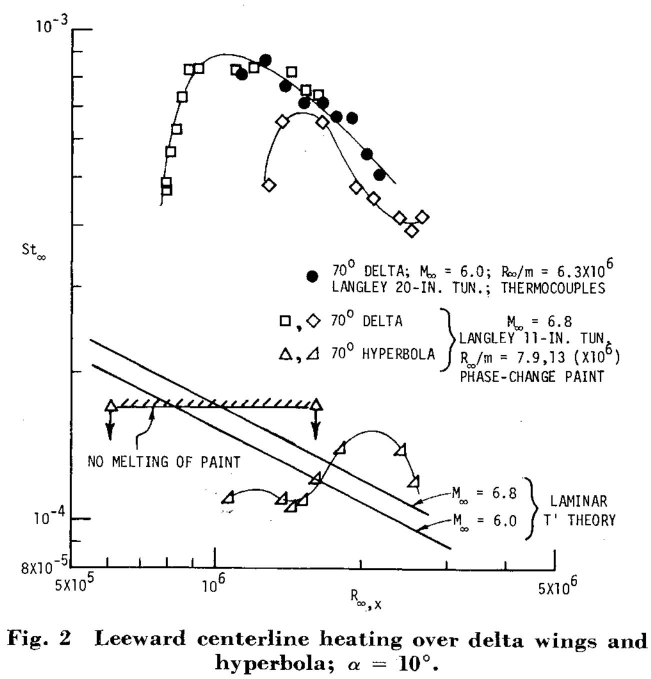

Fig. 2 Leeward centerline heating over delta wings and hyperbola; α = 10°.

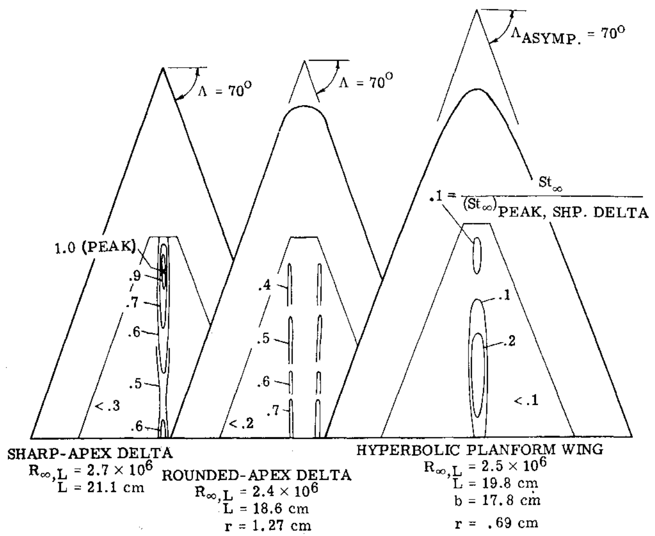

Fig. 3 Normalized heating contours; α = 10°; M_∞ = 6.8

The melt patterns on the hyperbola were difficult to interpret because of the low heat transfer and the lack of well-defined heating boundaries. However, under the interpretation which gives the highest heat-transfer values, these data fall on roughly the same level as laminar theory, or $\frac{1}{3}$–$\frac{1}{6}$ the heat-transfer level on the sharp-apex delta wing, despite the relatively small alteration made in the delta planform to obtain the hyperbolic planform shape (see Fig. 3). The heating alleviation obtained by contouring a wing leading edge must be related to the extent of the departure of the planform from the delta planform. For example, if the local sweep of the hyperbola approaches its asymptotic value too close to the apex, then the flow turns abruptly, which would probably result in vortex formation and centerline heating similar to that observed over the sharp-apex delta wings.

Heating contours shown in Fig. 3 have been normalized by the peak heating observed on the sharp-apex delta so all are directly comparable. The rounded-apex delta exhibits two narrow bands of relatively high heating coinciding with the high shear regions (Fig. 1). The heating within the data-gathering area of this configuration increases with distance from the nose in contrast to the heating distribution on the sharp-apex delta.

Vapor screen results, instrumental in constructing the lee surface flowfield about delta wings in hypersonic flow,[1,2,9] delineate the viscous boundaries over wing cross sections (Fig. 4). The top sketch in Fig. 4 illustrates a typical cross-sectional view of the lee side flowfield provided by the vapor screen. The photographs below are an enlargement of the flow over the center region of the models. The oil flow and the vapor screen for the sharp and rounded-apex deltas suggest the presence of paired vortices with axes located near the lo-

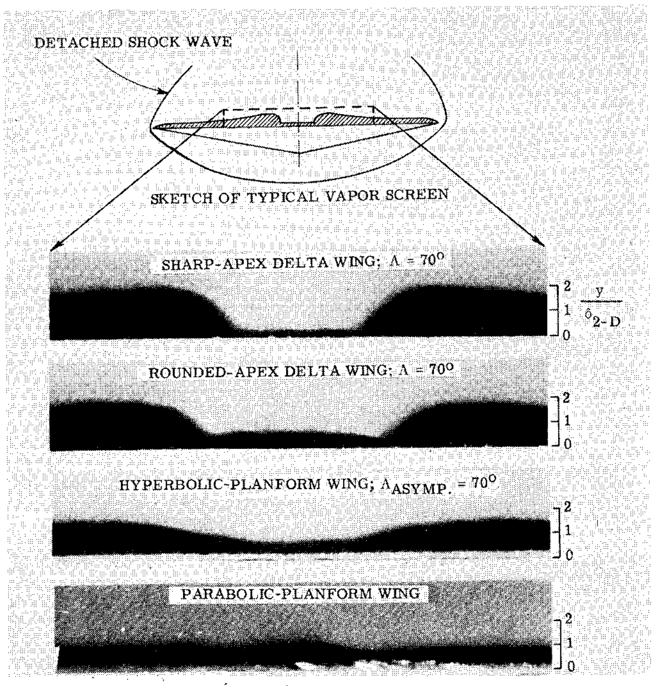

Fig. 4 Viscous boundaries in central region by vapor screen technique; $\alpha = 7°$; $x/L = 0.9$; $R_{\infty,x} = 9 \times 10^6$.

cation of maximum thickness of the viscous layer. This vertical motion induces a downward flow toward the centerline which then turns outward, drawing low energy fluid from the center area. The stagnating cross flow and the thinned boundary layer resulting from the departing fluid causes the increased heating to the centerline region. Highly thinned areas corresponding to the peak heating locations are observed in the central region on the sharp-apex delta and at symmetrically opposed locations off the centerline of the rounded apex delta. The relative magnitudes of the viscous-boundary height can be obtained from the scale provided, where the distance from the surface y has been normalized by the calculated laminar boundary-layer thickness[10] (δ_{2-D}) over a two-dimensional plate at 7° incidence in a cold flow. The minimum normalized viscous thickness for the wings exhibiting high, localized heating is roughly 0.3, whereas the minimum for the hyperbola and parabola, which show low heating and shear in the same region, is about 0.7. The vapor screen at the $x/L = 0.9$ station (x = distance down centerline from apex) of the hyperbolic wing suggests the initiation of the thinned central region characteristic of the delta wing flowfield. Not shown is an upstream vapor screen at $x/L = 0.4$ on the hyperbolic wing, which reveals a nearly continuous viscous boundary across the span with only a slight depression at the center. Only at a station much closer to the apex was a similar result observed on the sharp-apex delta wing.

The present results show the feasibility of reducing the vortex-induced heating to the lee surface of slender wings by properly contouring the leading-edge planform. The possibility of reducing upper surface heating to other configurations, such as the space shuttle, by proper contouring of the apex region has yet to be explored. Since leeward heating peaks on a current shuttle configuration have been reported to be of the same magnitude as on the compression side at angle of attack,[6] this problem deserves further study.

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Measurements of the Duration of Constant Reflected-Shock Temperature in a Reflected-Shock Tunnel

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Many studies[1–4] have shown that for shock tunnels operated in the overtailored condition, the duration of constant temperature can be considerably less than the duration of constant pressure behind the reflected shock. Davies[1] postulated a mechanism for this shortening, due to bifurcation of the reflected shock resulting from its interaction with the wall boundary layer. This allows early penetration of driver gas to the end wall above a certain incident shock Mach number. Markstein[5] applied a stability analysis to the problem and predicted breakdown of the contact surface and subsequent mixing of the driver and test gases above a certain Mach number.

In a recent series of chemical kinetic studies in a 6-in.-diam, 43-ft-long shock tunnel using a cold helium driver and a test gas of 10% carbon dioxide, 40% nitrogen, and 50% helium, it was necessary to operate in the overtailored condition in order to achieve the required reflected-shock temperatures. In order to establish the duration of contamination-free test time, an attempt was made to measure the reflected-shock temperature in the tube by monitoring radiation from the infrared active CO_2 in the test gas ahead of the end wall. It was felt that if driver gas reached the end wall, the resulting dilution and cooling of the test gas would decrease the radiation. An alternative procedure is that of Bull and Edwards[4] who added CO_2 to the driver gas and looked for the first indication of emission or absorption ahead of the end wall. Their data, however, indicated some erratic behavior, since with the CO_2 in the driver gas, dilution and cooling of the test gas have opposite effects on the radiation.

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