Leading-Edge Separations and Cross-Flow Shocks on Delta Wings

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Recently published numerical solutions of the Euler equations have shown the existence of very strong cross-flow shocks in the attached flow on the lee surface of uncambered delta wings. These shocks occur in flight conditions where previous work has shown that the lee-surface flow would be expected to show a leading-edge separation with a pair of contrarotating vortices inboard. This paper presents the results of an experimental study of the flow over one of the wings (mounted on a small body) for which numerical solutions were obtained. At the test Mach numbers where strong shocks occur in the numerical solutions, the experimental flow shows a complete leading-edge separation with a pressure distribution very different from that calculated. The significance of these results in relation to the change from leading-edge separation to attached flow around the leading edge with increasing Mach number is discussed.

Nomenclature

C_p = pressure coefficient
M = Mach number
M_n = Mach number normal to edge
η = spanwise distance from centerline normalized by local semispan
α_n = incidence normal to edge

Introduction

THE separated flow over the leeward surface of delta wings at supersonic speeds has been of interest for more than 30 years. Early research showed that on wings with leading edges swept well within the Mach cone from the wing apex, the lee-surface flow was similar to that at low speeds; that is, the flow separated from the leading edges to form a pair of contrarotating vortices about the wing. However, as Mach number increased, or sweep decreased, the vortices became flatter and eventually disappeared giving way to attached flow at the leading edge. Using all of the results available up to 1962, Stanbrook and Squire[1] found that the change from leading-edge separation to attached flow could be correlated in terms of the Mach number and the incidence normal to the leading edges, and showed that in these coordinates the two types of flow were separated by a band running from near $M_n = 0.6$ at low incidence to $M_n = 1.0$ at $\alpha_n \sim 30$ deg (Fig. 1). Basically to the left of the band the flow was always separated, whereas to the right of the band the flow was always attached at the leading edge, although shock-induced separations might occur inboard of the edges. Within the band both types of flow were observed and it was suggested that the width of the band was associated with the effects of Reynolds number and/or leading-edge radius; however, with hindsight it appears that much of the uncertainty in this region is due to the difficulty of determining whether a given flow is separated or attached at the leading edge, particularly when using published data.

Since 1964 several authors[2-5] have produced figures showing various additional flow regimes over the lee surface, and in the latest of these papers Szodruch and Peake[5] identify at least six different types of flow containing various combinations of contrarotating vortices and shock waves. Essentially most of the additional types of flow occur to the right of the Stanbrook-Squire boundary where the flow is attached at the leading edge. This attached flow expands supersonically around the edge giving a flow deflection toward the center. However, conditions of symmetry require the flow to turn eventually into the stream direction, and the resultant compression associated with this turning is often in the form of a cross-flow shock that may be strong enough to separate the boundary layer. On highly swept wings the separated flow then rolls up into a pair of contrarotating vortices. Clearly the physics of the separated flow with shock waves is easy to understand, but, to the author's knowledge, no completely satisfactory explanation for the change in flow pattern across the Stanbrook-Squire boundary has been given, although it must be associated with the formation of a cross-flow sonic surface between the leading edge and the shock wave under the wing surface so that supersonic expansion around the edge can take place.

Recently, there have been a number of numerical solutions for the flow over delta wings at incidence in a supersonic stream. In particular, Siclari[6] has used a finite difference scheme with an explicit shock-fitting routine to obtain solutions of the Euler equations and has calculated the flowfield over very thin elliptic cones. Some of these cones have cross sections with major-to-minor axes ratios as high as 14:1 so that the flow over them should resemble that over thin, flat, delta wings. All of the computed results show a very strong supersonic expansion around the edge onto the upper surface with the expansion region terminated by a cross-flow shock. A study of these results shows that in conditions which correspond to positions to the left of the Stanbrook-Squire boundary (i.e., the leading-edge separation region) the pressure ratio across the cross-flow is of order 3:1, whereas to the right of the boundary the ratio is much lower.

Clearly these inviscid results will be strongly affected by viscous effects, and these viscous effects would be expected to completely dominate the flow when the inviscid pressure ratio is as high as 3:1, thus it might appear that a further study of the elliptic cones investigated by Siclari might help in our understanding of the change in flow pattern from separated to attached flow at the leading edge. To this end, this paper presents the results of an experimental investigation of the flow over a thin elliptic cone identical to one of those for which Siclari has obtained numerical solutions.

Details of Experimental Investigation

The model chosen for study consisted of an elliptic cone with a semi-vertex angle of 20 deg in planview and a major-to-minor axes ratio of 13:90:1 (tan 20 deg:tan 1.5 deg) in cross

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section. Siclari has presented results for this case at Mach numbers of 2.0, 3.0, 4.0, and 6.0 at an incidence of 10 deg. The test model was made of steel and was 105 mm long, and in order to support the model in the tunnel, and to add strength, a circular cone with a base diameter of 12.7 mm was added symmetrically over the entire length at the center span. Tests were made in the intermittent blowdown tunnel in the Cambridge University Engineering Department. This tunnel has a rectangular test section with fixed nozzle blocks giving Mach numbers of 1.8, 2.5, and 3.5 with corresponding Reynolds numbers of 2.9, 3.2, and 5.2×10^7/m respectively. All of the tests were made with free transition. The model was mounted in the tunnel in various ways to obtain both planview and sideview schlieren photographs together with surface oil patterns on the lee surface. The model was then equipped with a small number of pressure tubes and pressures were measured across the span at 73 and 85 mm from the apex. Based on previous measurements on similar models in the same tunnel,[7,8] it is estimated that the pressure coefficients quoted here are accurate ± 0.006.

Presentation and Discussion of Results

Planview schlieren photographs at Mach numbers of 1.8, 2.5, and 3.5 for 10 deg incidence are presented in Fig. 2, while surface oil-flow patterns for various incidences at 1.8 and 3.5 are presented in Figs. 3 and 4, respectively. The sideview schlieren photographs only showed traces of the conical shock below the wing and are not presented here; however, the shock positions obtained from the photographs and from Fig. 2 are summarized in Fig. 5. The measured pressures on the upper and lower surfaces at $\alpha = 10$ deg at $M = 1.8$, 2.5, and 3.5 are shown in Fig. 6. In this figure, the dotted lines correspond to the numerical solutions of Siclari where these values were obtained by interpolation or extrapolation from the graphical results presented in Ref. 6, for Mach numbers of 2, 3, 4, and 6.

Considering the pressures on the lower surface (Fig. 6), first it will be seen that the measured pressures are always slightly higher than the calculated values. This increase is similar to the increase caused by a central conical body as calculated by Siclari (see Fig. 13 of Ref. 6).

Turning to the lee surface, it is clear that the measured and calculated pressures at $M = 1.8$ are completely different and are probably associated with different flow regimes. Essentially, the calculated pressure in the expansion field around the leading edge is much lower than the measured pressure, and the calculated shock is much closer to the leading edge than the measured pressure jump. This cannot be associated with the presence of the body since this would move the shock away from the centerline. (See Ref. 6, Fig. 13.) In addition, although the measured pressure jump is much less than that across the calculated shock it still corresponds to a pressure ratio of 2:1, yet there is no sign of a shock wave downstream of the trailing edge in the planview schlieren photograph (Fig. 2) although there is a dark region which may represent a vortex. Reference to the corresponding oil-flow photographs (Fig. 3) shows that the flow has separated from the edge to form the characteristic trace of a pair of contrarotating vortices above the wing. From this pattern it can be deduced that the flow over the vortices reattaches to the surface at $\eta = 0.45$ (i.e., just inboard of the compression), while the flow under the vortex separates in a secondary separation line at $\eta = 0.8$. The overall shape of the measured pressure distribution at $M = 1.8$ is very similar to that observed at low

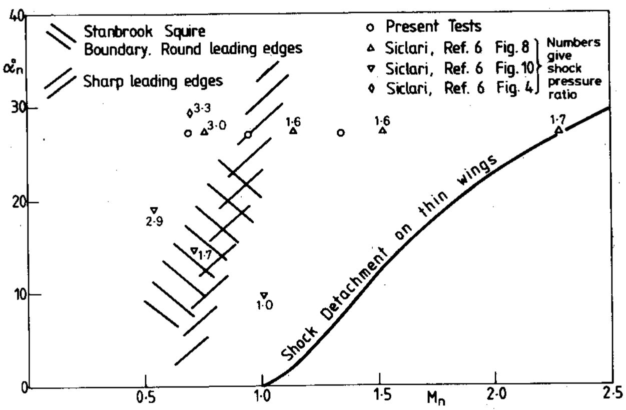

Fig. 1 Position of the Stanbrook-Squire boundary in relation to the present experimental results and the calculations of Siclari.

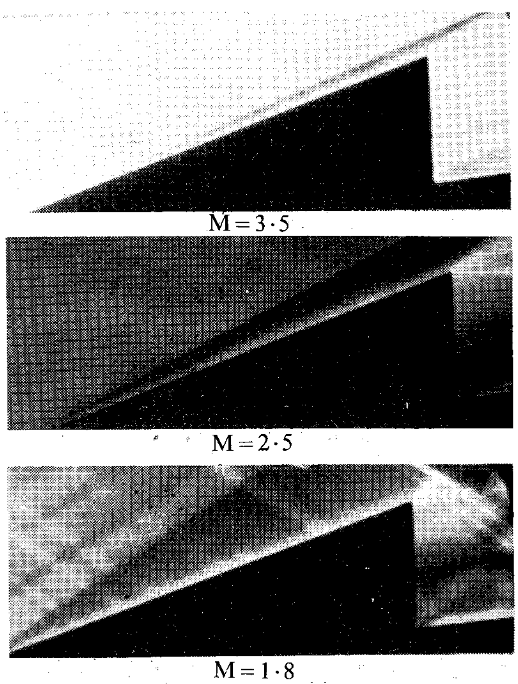

Fig. 2 Planview schlieren photographs.

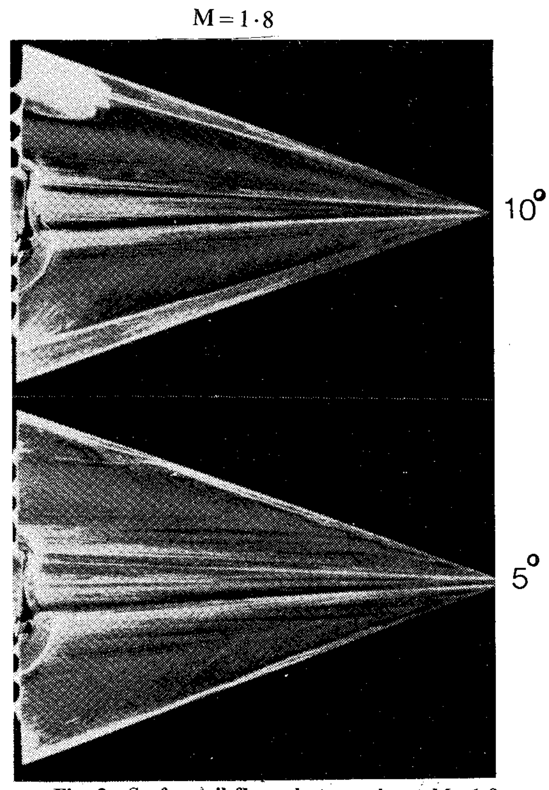

Fig. 3 Surface oil-flow photographs at $M = 1.8$.

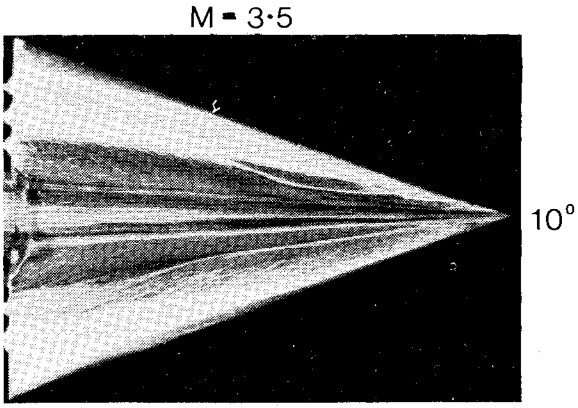

Fig. 4 Surface oil-flow photograph at $M = 3.5$.

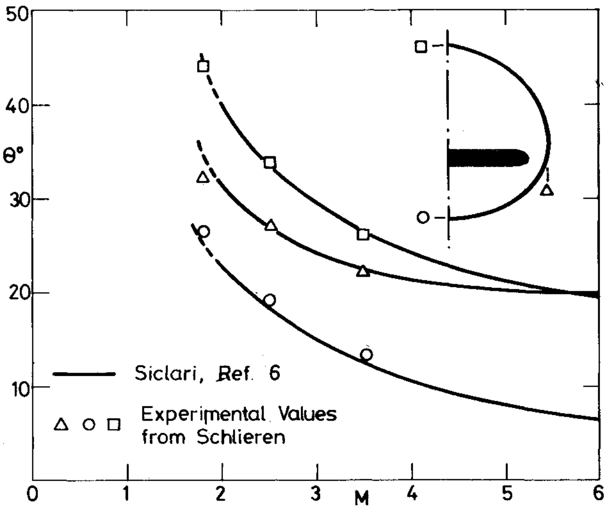

Fig. 5 Summary of measured and calculated shock positions.

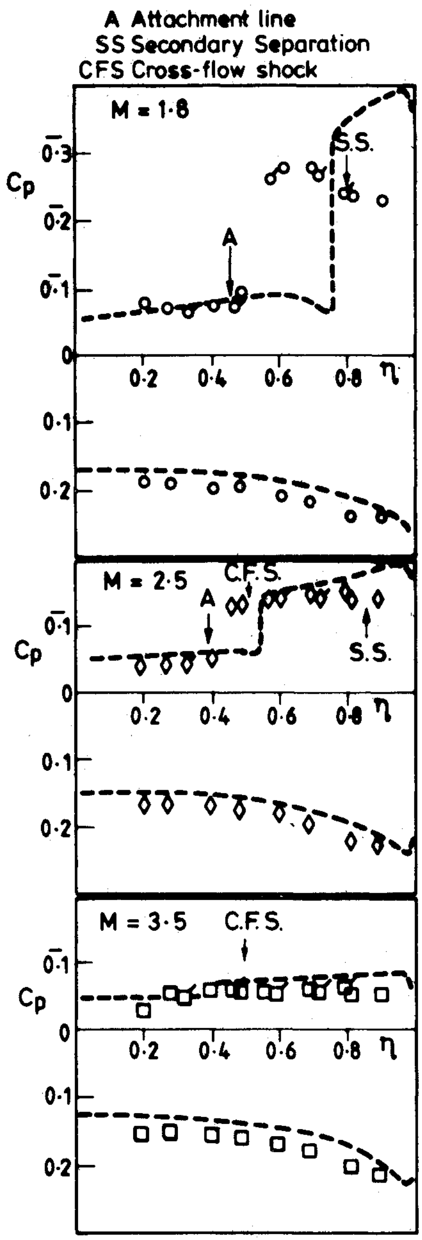

Fig. 6 Measured pressure distributions.

speeds, suggesting that the high peak suction at $\eta = 0.65$ is associated with the high spanwise and axial velocity directly under the vortex core. However, it should be noted that, according to this interpretation, the cross-flow velocity is directed away from the centerline and, therefore, the measured jump in pressure actually corresponds to an expansion from the high pressure near the attachment line to the suction peak associated with the high cross-flow and axial velocities under the vortex core. The presence of secondary separation close to the edge reinforces this view of the flow since it occurs as the crossflow encounters an adverse pressure gradient as the suction decreases away from the vortex core. A sketch of this flow is shown in Fig. 7, where some of the features are based on published vapor-screen photographs of the vortex flow over wings at similar values of α_n and M_n to those of the present test. For example, the vortex sheet leaving the leading edge, the vortex core, and the secondary separation, are all clearly shown in Fig. 30 of Ref. 9. (Unfortunately, it has not been possible to obtain vapor-screen photographs in the small tunnels used in the present investigation.) It should also be noted that the surface oil-flow pattern at $\alpha = 5$ deg is also typical of that associated with leading-edge separation so that the vortex flow at $\alpha = 10$ deg is well established. In spite of this, it is interesting to note that the shape of the shock wave below and around the wing is almost identical to that calculated by Siclari, as is shown in Fig. 5. In fact, this agreement between the measured and calculated shock shape is seen at all the test Mach numbers.

At $M = 2.5$, the measured and calculated pressure distributions are in much closer agreement, but again the measured pressure rise is further inboard than the calculated shock position, and the measured pressure just inboard of the edge is slightly higher than that in the calculated expansion region. As at $M = 1.8$, the surface oil pattern (not presented here) is characteristic of leading-edge separation and the positions of the attached and secondary separation lines are marked on the pressure distribution in Fig. 6. The planview schlieren photograph for this Mach number shows a conical line emanating from the trailing edge at about half the semispan. This line is assumed to be the trace of the cross-flow shock (CFS); its position is also included in Fig. 6. From Fig. 6 it will be seen that, as at $M = 1.8$, the attachment line is just inboard of the main compression, whereas the cross-flow shock is just outboard; the pressure in the expansion region is also almost constant. Together, these facts suggest that the separated flow over the wing is now much shallower and the circulation around the vortex weaker than at $M = 1.8$, and that the main feature of the flow is now a supersonic expansion terminated by a cross-flow shock on top of the separated flow. A sketch of this flow is also shown in Fig. 7 where some of the features are based on vapor-screen

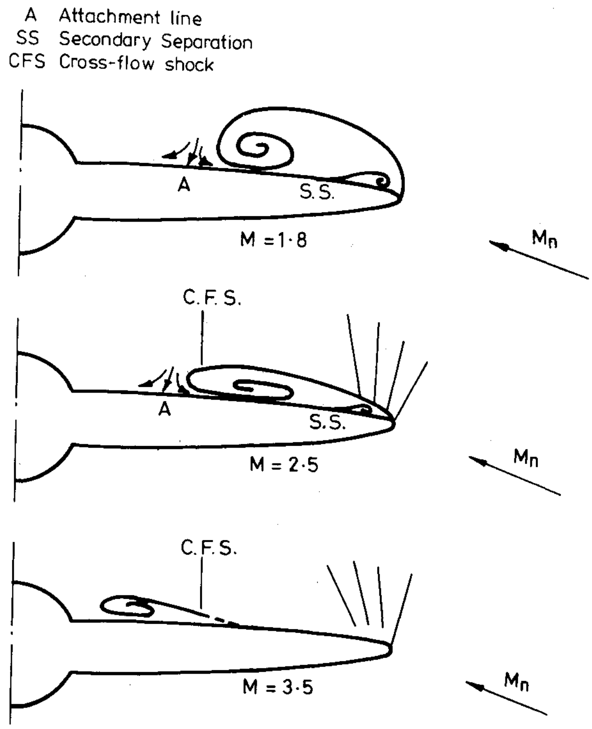

Fig. 7 Sketches of the flow patterns above the wing at the test Mach numbers.

photographs of the flow over delta wings of 65 deg sweep in this Mach number range, as tested by Squire et al.[10]

The measured pressure distribution on the suction surface at $M=3.5$ is almost flat, although there is a possible compression just inboard of the calculated value and, again, the measured pressure near the edge is slightly higher than the calculated value. Also at this Mach number there is a clear trace of the cross-flow shock downstream of the trailing edge in the planview schlieren at $\eta=0.5$. On the other hand, the oil flow is less clear; the flow appears to be attached around the leading edge with an oil accumulation line at $\eta=0.9$ and a fairly definite pattern of a vortex flow further inboard. This type of pattern is similar to that found by Mason and Miller[11] over a 57 deg swept delta wing at $M=1.7$ and by Squire et al.[10] on a 65 deg swept wing at $M=1.8$ and 2.0; i.e., in conditions just to the right of the Stanbrook-Squire boundary. In particular, all of the published photographs of surface oil flow in these conditions show a clear accumulation of oil just inboard of the leading edge. As pointed out by Mason and Miller, and confirmed by the present tests, this oil accumulation occurs in a region of constant, or favorable pressure gradient, so it is not caused by separation. On the other hand, it does occur in a region where the shear stress is falling rapidly from the very high laminar values on the stagnation streamline and then rising again as the layer becomes turbulent. As shown in calculations[12] for the motion of an oil sheet under a boundary layer, this type of shear distribution can cause an accumulation of oil resembling that at separation. This interpretation is supported by the fact that at lower incidences the oil accumulation line only occurs near the nose and further back over the wing the oil streak lines are completely typical of an attached flow around the leading edge with the flow inboard compressing and then turning into the freestream direction.

The main features of the flow at $M=3.5$ at $\alpha=10$ deg are also sketched in Fig. 7, where it should be noted that the position of the separation associated with the cross-flow shock wave cannot be obtained with certainty, but it clearly must be outboard of both the cross-flow shock and the edgewise flow under the vortex. Further evidence for these features can be found in the oil-flow photographs and

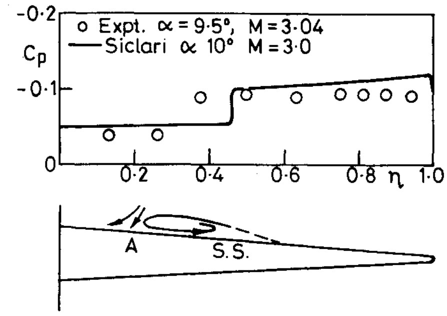

Fig. 8 Experimental results of Thomann.[13]

pressure distributions obtained by Thomann[13] at a Mach number of 3.0. In particular, the oil-flow results obtained by Thomann on a 65.9 deg swept delta wing at an incidence of 9.5 deg are very similar to those obtained in the present investigation at an incidence of 7.5 deg at $M=3.5$. Thomann also obtained results on a 70 deg swept delta wing at an incidence of 9.5 deg at $M=3.0$, which is very close to one of Siclari's calculated results. The measured and calculated pressure are compared in Fig. 8. As seen from the figure, the test wing does not have an elliptic cross section; therefore, exact agreement between the measured and calculated results should not be expected. However, the results presented in Fig. 8 again show that the position of the measured pressure rise is outboard of the attachment line as shown by the surface oil flow and is inboard of the cross-flow shock as predicted by the numerical results.

Taken together these results show that up to a Mach number of 3.5 at least the flow over the upper surface of the wing is dominated by viscous effects in the form of rolled up vortices. Particularly at $M=1.8$ and 2.5, the sharp pressure change at the inboard edge of the low-pressure region near the leading edge appears to be closely associated with the reattachment of the external flow just inboard of the vortex system with no evidence that the cross-flow shock which terminates the supersonic expansion around the edge in inviscid flow actually reaches the surface. Of course, the present results and those of Refs. 10, 11, and 13 were obtained at relatively low Reynolds numbers so that in the present tests, for example, the flow near the apex of the wing was probably laminar, as suggested by the change in the flow pattern just ahead of midchord. However, this appears to be a change in the scale of the separation rather than a change in the nature of the separated flow. This interpretation is supported by the oil-flow photographs of Ref. 10 with fixed and free transition which show that the flow patterns (and overall forces) with leading-edge separation are completely independent of transition fixing. However, when the flow is attached at the leading edge transition does change the size of the separation under the shock and the pressure distribution on the region of the interaction as shown, for example, by the results of Mason and Miller.[11] Similar conclusions are reached by Szodruch[14] who has made a careful study of the effects of Reynolds number on these types of flow.

Concluding Remarks

Although the experimental results presented herein show that viscous and vortex effects are very dominant on the upper surface, it is clear that at the highest test Mach number (3.5) these viscous effects have only a small influence on the pressure distributions as compared with the theoretical inviscid results. Whereas at the lowest test Mach number of 1.8 the viscous/inviscid interaction produces a complete change from the inviscid flow pattern with the flow completely separated from the leading edge. Essentially it would appear that in cases where the inviscid solution predicts pressure ratios across the shock close to, and just above, that necessary to separate the turbulent boundary layer (i.e., on the order of

1.6:1), then the boundary layer separates just ahead of the inviscid shock position and the resultant vortex system modifies the pressure distribution slightly and, in particular, moves the pressure rise inboard. As the Mach number falls and the pressure ratio across the inviscid shock rises, the separation occurs further and further outboard of the inviscid shock position and the vortex system grows in size and strength, thus producing a greater change from the inviscid pressure distribution. Eventually the separation line reaches the leading edge and the overall pressure distribution bears little resemblance to that predicted by the inviscid solution.

Clearly, it is unlikely that the first appearance of leading-edge separation will depend solely on the magnitude of the pressure ratio across the cross-flow shock as calculated by inviscid theory. However, it is equally clear that if this pressure ratio is significantly greater than that required to separate the turbulent boundary layer, then extensive separation will occur under the shock with consequent large changes in the pressure distribution as shown by Korkegi.[15] Thus, computed pressure ratios greater than about 2:1 should be taken as an indication that inviscid theory is likely to be inacccurate and leading-edge separation may occur at slightly lower Mach numbers or higher incidences.

Although all of the experimental results used in this paper were obtained at low Reynolds numbers compared with those in flight, regions of extensive separation with vortices under strong swept shocks have been observed at higher Reynolds numbers.[15] Thus, the mechanism for the change to leading-edge separation discussed above should apply at full-scale flight conditions. In this connection, it should be noted that when inviscid theory is used to design cambered wings with weak cross-flow shocks then extensive separation should not occur, but the present results do suggest limits on the strength of the cross-flow shocks allowable in the design process.

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