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Separation Ahead of Steps on Swept Wings

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Introduction

THE character of the boundary layer is one of the most important factors influencing separation; laminar boundary layers separate more readily and more extensively than turbulent boundary layers.[1,2] Many investigations have been made of two-dimensional laminar, transitional or turbulent boundary-layer separation, but few have examined the effects of boundary-layer transition on three-dimensional separated flows. Korkegi[3] described the interaction of a planar swept shock wave with the boundary layer on an unswept flat plate. The present work addresses the complementary problem of a shock caused by an unswept obstacle interacting with the boundary layer on a swept flat plate.

Theoretical Analysis

Whitehead and Keyes[4], Stollery[5] and others have observed "dumbbell" shaped regions of separated flow ahead of trailing edge flaps on delta wings, such as that sketched in Fig. 1. The shape of this region, over half the planform, reminds one of plots showing the variation of separation length, ℓ, with Reynolds number (e.g., Ref. 4). For small Reynolds numbers the boundary layer remains laminar through reattachment, and the extent of separation increases with increasing Reynolds number. This trend is reversed (for larger Reynolds number, ℓ decreases with increasing Reynolds number) when transition occurs in the separated shear layer prior to reattachment. At still larger Reynolds numbers, transition occurs upstream of separation, and the extent of separation becomes insensitive to further increases in Reynolds number.

Boundary-layer transition on sharp-leading-edge swept wings at zero angle of attack occurs in a region parallel to the wing leading edge, and, in the absence of protuberances or other disturbances, flow over the wing remains very nearly streamwise.[4] Thus, considering a strip-type analysis, turbulent separation could occur inboard while outboard the separation could be laminar. In between, the separated flow would be characteristically transitional (cf., Fig. 2.).

Although the motivation here is to understand separation ahead of flaps or elevons on swept wings, the flow phenomenon can be understood more readily by considering separation ahead of forward facing steps on swept wings. Reattachment occurs close to the step shoulder.[6,7] The length

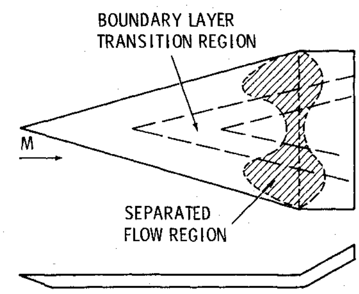

Fig. 1 "Dumbbell" shaped region of separated flow (Ref. 5).

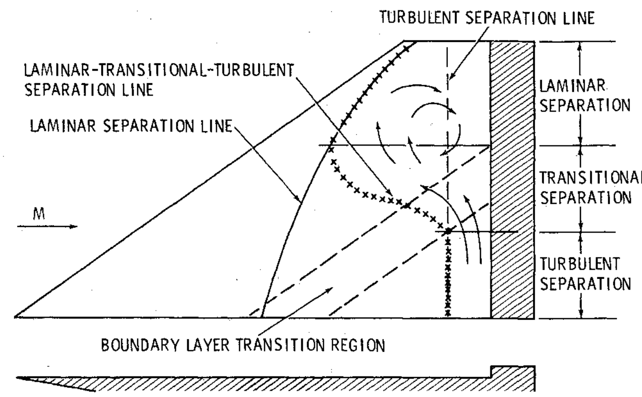

Fig. 2 Separation line shapes on a swept wing ahead of a forward facing step.

of separation ahead of the step can be estimated by using the step height h and assuming a straight dividing streamline. The angle of the dividing streamline α is estimated using oblique shock relation[8] for the average pressure level, $P = p_p/p_1$, on the plate surface in the separated flow region ahead of the step. Thus[8]

$$\ell = \frac{h}{\tan\alpha} = h\left[\frac{7M^2-5(P-1)}{5(P-1)}\right]\sqrt{\frac{6P+1}{7M^2-(6P+1)}} \quad (1)$$

where M is the undisturbed flow Mach number.

For two-dimensional separated flows ahead of steps, empirical expressions have been well established for the average pressure rise P for either laminar[9]

$$P = 1 + 1.22M^2\left[(M^2-1)Re\right]^{-1/4} \quad (2)$$

or turbulent[10]

$$P = \begin{cases} 1 + 2.24M^2 \Big/ \left[8 + (M-1)^2\right] & \text{for } M < 3.4 \\ 0.091M^2 - 0.05 + 6.37/M & \text{for } M > 3.4 \end{cases} \quad (3)$$

separation. For turbulent separation, the average pressure rise in the separated flow region can be estimated as a function of M alone [Eq. (3)]. Knowing M, and hence P from Eq. (3), the extent of turbulent separation can be estimated using Eq. (1). The separation line calculated for fully turbulent boundary-layer flows for a sample case is indicated by the dashed line in Fig. 2. For laminar separation, P is a function of both M and also the Reynolds number, Re, based on local flow conditions and the streamwise distance from the wing leading edge to the separation location. In the laminar case, Eqs. (1) and (2) must be solved iteratively to yield a separation locus such as that indicated by the solid curved line in Fig. 2.

If there are substantial regions of both laminar and turbulent flow on the wing surface, and the proposed strip

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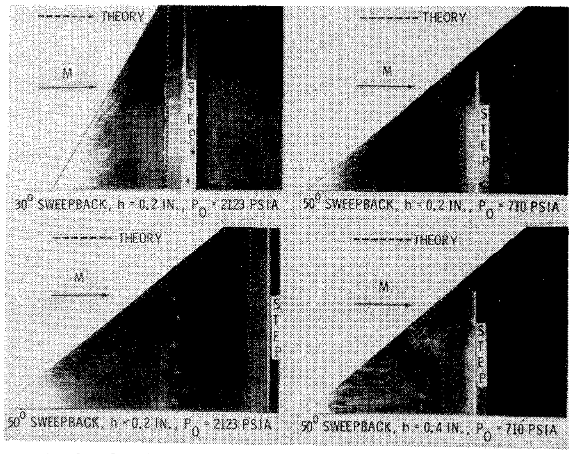

Fig. 3 Oil flow patterns ahead of steps on swept wings.

analysis is valid, then the actual separation line should resemble that indicated by the x's in Fig. 2. On the inboard portion of the wing, separation will occur along the turbulent line; on the outboard portion separation will occur along the laminar line. In between, the separated flow will be transitional, and the separation line will follow an "s" shaped curve such as that indicated in Fig. 2.

The existence of standing vortices, crossflow, and premature transition in a free shear layer make suspect the use of a two-dimensional strip-type analysis. However, these are secondary effects and do not invalidate the postulated method as a means for estimating qualitatively the effects of transition on the extent of separation ahead of controls on swept wings.

Experimental Program

Experiments were conducted in the High Reynolds Number Aerospace Research Lab, Mach 6 Tunnel. To ensure reasonable extents of both laminar and turbulent boundary-layer flows, it was opted to use "half wings" rather than full delta wings. Wings with machined sharp leading edges with sweepback angles of 0, 30, and 50° were fabricated. Steps of two different heights ($h = 0.2$ and 0.4 in.) were provided. The steps could be mounted either at the mid-chord or at the aft end of the wings. The steps are sealed to the wings, to prevent flow between the step and wing surface. The step heights are comparable to or larger than the undisturbed boundary-layer thickness along the inboard portions of the wings.

Oil flow and schlieren photographs were obtained for both step heights at both locations on the three wings for two tunnel flow stagnation pressure levels: approximately 710 and 2123 psia. The resulting freestream unit Reynolds numbers are approximately 9.4 and 27.2 million per foot (approximately 12 and 34 million based on the 50° wing root chord); the freestream Mach number is approximately 5.88 for both pressure levels.[11]

Sample oil flow photographs are shown in Fig. 3. At the higher pressure level, the flow over the wing was predominately turbulent and the oil accumulation lines indicate nearly a constant extent of separation ahead of the steps. For the lower pressure level, the oil accumulation lines are similar to the x line sketched in Fig. 2 for turbulent separation inboard, laminar separation outboard, and transitional separtion in between.

Comparison of Data with Theory

For turbulent separation, the measured oil accumulation lines indicated an average separation length of approximately $5.2h$. The theoretical value, indicated by the dashed lines in Fig. 3, is $4.9h$. We believe that the small discrepancy is caused by the subject step heights being comparable to the boundary-layer thicknesses on the plate surface.[6] For laminar-transitional-turbulent separation, the photographed oil accumulation lines closely resemble those predicted (dashed lines in Fig. 3) using the theoretical analysis previously described.

Conclusions

The postulated strip-type analysis correctly predicts, qualitatively, the shape of separated flow regions ahead of forward facing steps on swept wings. The proposed method is quatitatively correct, at least within the range of conditions for the subject experiments, as long as the location of boundary-layer transition on the swept wings is estimated correctly.

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Explicit Equations for Barometric Altitude Computations

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THE values of barometric altitude, Mach number, and various airspeeds are calculated by onboard air-data systems from measured pressures and temperatures. The equations that relate the measured values to the required quantities, although simple, involve noninteger exponents. But in supersonic range, the Mach number and airspeeds are related to measured values in an implicit fashion, thus requiring iterations for implementation in the onboard computer. To avoid such iterations, Bogel[1] has obtained explicit expressions for computation of supersonic Mach numbers and airspeeds. The expressions he obtained involve only square root and summations operations. Bogel also has im-

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