base pressures were measured by averaging the static pressure readings of four static tubes at the base, connected to an inclined (30°) alcohol manometer. The measurements quoted are the averages obtained at three speeds (typically 70, 90 and 100 fps).

Results and Discussions

Figure 2 shows the variation of C_D vs $C_D(S/C)$ from the present and the earlier data. It is observed that this relation tends to be a straight line within experimental scatter up to the maximum blockage tried, regardless of the nearness of one wall and the geometric shape of the body. The slope of the straight line fitted here is slightly less than the value given by Maskell on the basis of data for up to only 5% blockage but the linear relation seems to be hardly in doubt. This relation in general can be written as

$$C_D = C_{D_c} + 0.8 C_{D_c} C_D(S/C)/K_{cc}^2 \qquad (2)$$

It is also observed that C_{Dc} is practically independent of the distance from the wall but that K_c^2 changes significantly as the wall is reached. This variation is shown in Fig. 3 where $[k_c^2 + C_{pb}]$ is plotted against $k^2(S/C)$. The relation is observed to be an exponential of the form

$$C_{p_b} = -k_c^2 + \exp(-A_2 k^2 (S/C)) \qquad (3)$$

It is obvious that for small (S/C) at the center of the wind tunnel, Eq. (3) must be compatible with Eq. (2) with (C_D/k^2) constant. A further check in fact shows that A_2 is very closely 0.8 C_{Dc}/K_c^2 so that

$$C_{p_b} = -k_c^2 + \exp(-0.8 C_{D_c} k^2 (S/C)/K_c^2) \qquad (4)$$

Assuming the similarity of the parameter $(p - p_b(1)/(H - p_b)$ under constraint, Maskell showed that a straightforward application of momentum balance yields, with a few plausible assumptions, a linear relation between C_D and $C_D(S/C)$ for small (S/C). The fact that this is valid for blockages up to 20% indicates that the significant local departure of pressure at a point, from the similarity, tends to even out in the integrated pressure differential.

The quantity $[C_{Dc}/K_c^2]$ was found to be independent of the ratio of model size to boundary-layer thickness (about 2 in.) in our experiments, depending only on the fraction of the boundary-layer thickness at which the bottom edge of the plate was located. Closer to the wall and at the wall, when an upstream bubble was presumably formed (Fig. 1), Eq. (3) could be described by the simpler relation:.

$$C_{p_b} = C_{p_{bc}} + (C_{D_c}/K_c^4) C_{p_b}(S/C) \qquad (5)$$

It is therefore observed that the base pressure variation depends on the three regimes namely the uniform upstream, shear flow, and proximity to wall. A plot of C_D/k^2 and C_D/K^2 is shown in Fig. 4 where it is observed that the range of (S/C) over which Maskell's Hypothesis is valid seems to decrease with increased nearness to the wall as would be implied also from Eqs. (2) and (4). On the other hand, the quantity C_D/K^2 tends to a constant value for all (S/C) tried as the wall is reached. Thus it is observed that while far from the wall, the hypothesis that C_D/k^2 is constant is valid for small S/C, $C_D/(k^2 - 1)$ is constant at and very close to the wall. The latter condition implies that very near and at the wall $(p - p_b)/(p_\infty - p_b)$ remains invariant under constraint. Clearly the velocity scale is not the shear layer velocity kU_0 found by Maskell far from the wall, but depends on the dynamic pressure differential $[((1/2)\rho U_0^2 - (1/2)\rho k^2 U_0^2]$, K^2 being presumably related to the reattachment velocity.

Following Maskell,[1] one can derive the expression for drag coefficient of the body under constraint as

$$C_D = m\{k^2 - [C/(C-B)]\} \qquad (6)$$

[Maskell's Eq. (8) simplified]. From the condition that

$$C_D/K^2 = C_{D_c}/K_c^2 = m\{k^2 - [C/(C-B)]\}/K^2 = m_c \qquad (7)$$

one can derive for small S/C, assuming that $m = aK^2$,

$$C_D = C_{D_c} + (C_{D_c} C_D (S/C)/K_c^4) \qquad (8)$$

$$K^2 = K_c^2 - (C_{D_c} C_{p_b}(S/C)/K_c^4) \qquad (9)$$

In the present experiments, C_{Dc} 1.15 and $K_c^2 \approx 0.66$, giving $C_{Dc}/K_c^4 = 2.65$ compared with the experimentally observed value of 2.56. As Maskell has pointed out, some additional corrections due to wake distortion has to be made but this is not attempted; these corrections seem, in any case, to be small.

Application

To apply the corrections, one measures C_D and C_{pb} (*or*-K^2) and uses the pair of relations (8) and (9) for bodies on the wall, to evaluate C_{Dc} and C_{pbc} Far from the wall, Eqs. (2) and (4) can similarly be used. Elsewhere in the presence of shear, Maskell's procedure is observed to be insufficient to determine C_{Dc} and a further relation between K_c and K_{cc} is required.

References

[1] Maskell, E. C. "A Theory of the Blockage Effects on Bluff Bodies and Stalled Wings in a Closed Wind Tunnel," R&M 3400, 1965, Aeronautical Research Council, England.

[2] Melbourne, W. H. and McKeon, R. J., "Wind Tunnel Blockage Effects and Drag on Bluff Bodies in a Rough Wall Boundary Layer," Third International Conference on Wind Effects on Buildings and Structures, Vol. 1, Pt. II, 1971, Tokyo, pp. 9.1–9.9.

[3] Buckle, P. R. and Boak, A. G., "A Study of Blockage Effects of Bluff Bodies in Closed Wind Tunnels," Rept. 117, 1968, Department of Aeronautical Engineering, University of Bristol, England.

[4] Modi, V. J. and Sherbiny, S. EL. "Effect of Wall Confinement on Aerodynamics of Stationary Circular Cylinders," Third International Conference on Wind Effects on Buildings and Structures, Vol. 1 Part II, 1971, Tokyo, pp. 19.1–19.11.

Prediction of Airfoil Shock Location in Transonic Flow

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IT is well known that a wing moving at transonic speeds will have a near normal shock wave standing near the midchord. Usually, this shock wave is coupled with a local boundary-layer separation. The shock wave location strongly influences[1] the aerodynamic force and thus the performance and stability characteristics of an aircraft.

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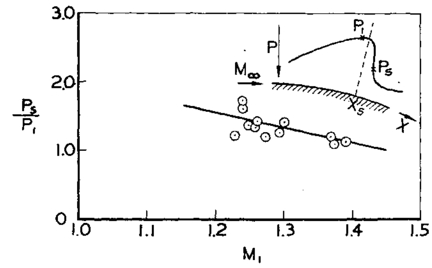

Fig. 1 Pressure ratio vs local onset Mach number.

Because of the nature of transonic inviscid/viscous flow interactions, there is neither a good theory nor an empirical method capable of predicting the flowfield for a significant interactions case.

Most inviscid flow theories do not accurately predict shock wave locations when the shock wave is associated with a boundary-layer separation. Thus the prediction of the shock wave movement along chordwise stations has become one of the most fascinating topics in recent years. The purpose of this Note is to provide an engineering method for predicting the shock wave location on an airfoil. The proposed procedure is simple, yet exhibits satisfactory agreement with wind-tunnel observations.

Mager[2] has demonstrated the existence of a correlation between the onset Mach number and the pressure ratio between the freestream static pressure and the static pressure at separation for a turbulent supersonic boundary-layer separation. Mager also suggests a correlation between the Mach number ahead of the shock wave and the local Mach number at separation. However, at transonic speeds, linearized theory, such as Mager's becomes doubtful. In this regard, empirical correlations, analogous to Mager's were established based on experimental observations.

A test program[3] was conducted on a C-141 airfoil in the Arnold Engineering Development Center 4-ft transonic wind tunnel. The model was a two-dimensional section with end-plates and had a 6-in. chord. The test was conducted at a Reynold's number of five million/foot. The boundary-layer separation point was established with a surface oil-flow technique. Based on this study, correlations of pressure ratio with the onset Mach number and with the local Mach number at separation were established (with the subscript indicating local value) and are shown in Figs. 1 and 2, respectively. Clearly, the surface curvature effect is included in these correlations. In general, the surface curvature of most airfoils is about the same. In view of the relatively weak influence of the curvature effect, it is proposed, for an engineering prediction, that these correlations can be extended to more general airfoil configurations.

These correlations must couple with a good transonic inviscid flow theory. For this study, Krupp's numerical solution was employed.[4] Figure 3 is a sample calculation for pressure coefficient, C_p, of the airfoil tested compared

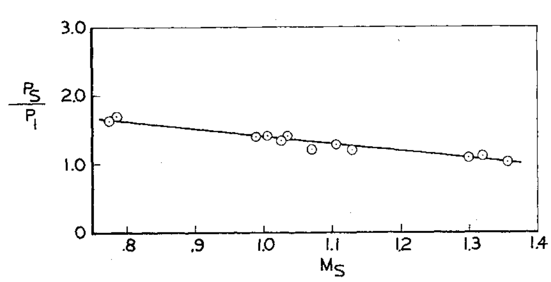

Fig. 2 Pressure ratio vs local separation Mach number.

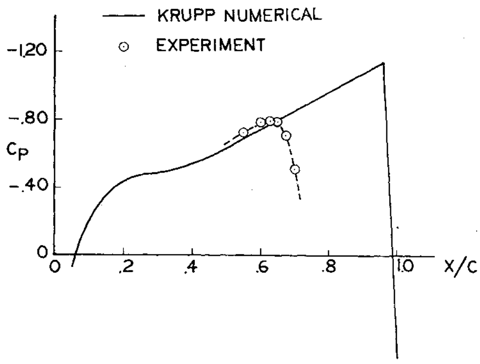

Fig. 3 Typical upper surface pressure distribution, $M_\infty = 0.90, \alpha = 0°$.

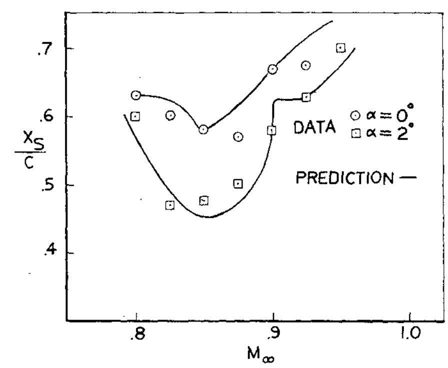

Fig. 4 Upper surface shock location vs flight Mach number.

with the experimental data plotted on normalized chordwise station x/c. As expected, the agreement with respect to the shock location is not good. Now assuming that the separation pressure occurs in the shock and calculating the corresponding local Mach number, the pressure ratio can be determined from Fig. 2. Going to Fig. 1 this pressure ratio now determines the onset Mach number. Locating this Mach number on the Krupp result, establishes the shock location. The results of this procedure are shown in Fig. 4, where x_s/c is the shock location on normalized chordwise station measured from the leading edge.

The correlation curves were developed for three freestream Mach numbers: 0.80, 0.85, and 0.90. The airfoil was held at a constant geometric angle of attack of zero degrees. To demonstrate the correlations are indeed applicable to a wider range, the procedure was employed in predicting the shock location at the intermediate Mach numbers as well as extrapolated to a freestream Mach number of 0.95. The prediction agrees well with the data. Also, the procedure was accurately extrapolated to an incidence angle (α) of 2°. For the Mach number of 0.80 case, the flow did not fully separate from the airfoil surface whereas the cases of 0.85 and 0.90 did fully separate aft of the shock. It is significant that this procedure predicted the forward motion of the shock with increasing Mach number which has never been predicted by any inviscid flow theories. The phenomena of forward motion of the shock is of utmost importance in interpreting[5] some peculiar performance and stability characteristics of aircraft operating at transonic speeds.

References

[1]Pearcey, H. H., Osborne, J., and Haines, A. B., "The Interaction Between Local Effects at the Shock and Rear Separation—A Source of Significant Scale Effects in Wind Tunnel Tests on Aer-

ofoils and Wings," *Transonic Aerodynamics*, AGARD Conference Proceedings 35, 1968, pp. 11-1 to 11-23.

[2]Mager, A., "On the Model of the Free, Shock-Separated, Turbulent Boundary Layer," *Journal of the Aeronautical Sciences*, Vol. 23, Feb. 1956, pp. 181–184.

[3]Smith, D. K., "Wind Tunnel Investigation of the Pressure Distribution on a Two-Dimensional Airfoil with Pylon Mounted Stores at Mach Numbers from 0.70 to 0.90," AEDC-TR-73-71, March 1973, Arnold Engineering Development Center, Arnold Air Force Station, Tenn.

[4]Krupp, J. A., "The Numerical Calculation of Plane Steady Transonic Flows Past Thin Lifting Airfoils," PhD. thesis, June 1971, University of Washington, Seattle, Wash.

[5]Studwell, V. E., "Investigation of Transonic Aerodynamic Phenomena for Wing Mounted External Stores," PhD. thesis, June 1973, University of Tennessee Space Institute, Tullahoma, Tenn.

Longitudinal Response of an Aircraft due to a Trailing Vortex Pair

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A survey of the available literature on prediction of airloads on aircraft due to trailing vortices of another aircraft and subsequent dynamics of the aircraft subjected to such loads revealed a limited amount of information. Research in this area has been limited due to the complexity in adequately describing the vortex formation, motion, and dissipation. Furthermore, flight test data would be almost impossible to obtain due to the apparent danger imposed on the aircraft and its crew.

In this study, an approximate method to predict the response of a small aircraft in longitudinal motion when it is flying through a trailing vortex system of a large aircraft is developed. In the analysis, the vortex system was idealized as two parallel line vortices in a horizontal plane. The following aircraft was in horizontal flight at a fixed distance from the plane of the vortices and did not perturb the vortices. Several other simplifying assumptions were made: 1) controls are fixed, 2) coupling effects between the lateral and longitudinal motions are neglected, 3) spinning rotor effects are neglected, and 4) effect of the aircraft on the trailing vortices is neglected.

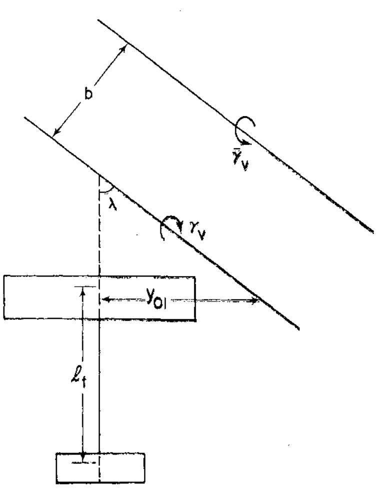

Fig. 1 Aircraft-vortex pair representation.

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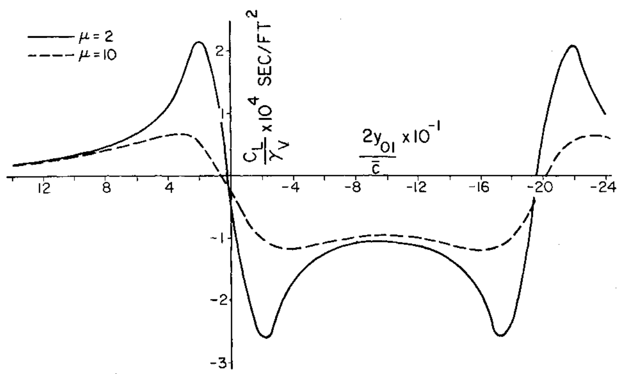

Fig. 2 Variation of incremental lift coefficient per unit vortex strength with position of left vortex.

Table 1 Dimensions and specifications of De Havilland Beaver

C.G. location	
Aft L.E. MAC	35.0% MAC
Below L.E. MAC	36.0 in.
Max gross weight for this condition	4800 lbs
Areas	
Wing, total	250 ft^2
Horizontal tail, total, approx.	43.82 ft^2
Dimensions and general data	
Wing span	48 ft
Horizontal tail span	15.83 ft
Wing chord (MAC)	5.2 ft
Horizontal tail chord (MAC), approx.	2.76 ft
Dihedral	2°
Wing aspect ratio	9.2
Tail aspect ratio, approx.	5.71
Distance from wing MAC quarter chord point to horizontal tail MAC quarter chord point (l_t)	19.76 ft

The longitudinal stability equations of motion were linearized as in Ref. 1 by the application of the small disturbance theory and the external aerodynamic loads were assumed to consist of two parts. The first part was due to the presence of the trailing vortex pair and the second part was due to the changes in the flight variables which are conventionally represented in terms of the stability derivatives. The analysis was applied to a particular small

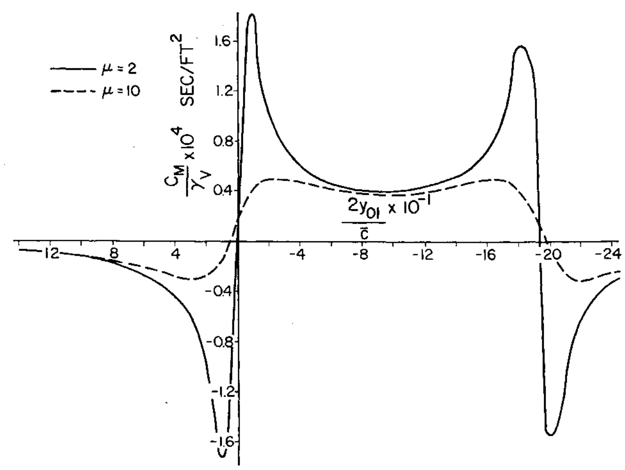

Fig. 3 Variation of incremental pitching moment coefficient per unit vortex strength with position of left vortex.