Engineering Notes

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Calculation of Forces on Stores in the Vicinity of Aircraft

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ONE of the principal tools of the theoretical aerodynamicist is Munk's theory of slender bodies.[1] The theory was originally developed for dirigibles, but it was revitalized by R. T. Jones who extended it to three-dimensional airfoils. This Note summarizes the results obtained by applying the theory to complex configurations (Fig. 1) such as an aircraft-store combination. The objective of the analysis was the calculation of the local velocity around the aircraft body and the subsequent calculation of the forces (except drag) and moments (except the rolling moment) on the store. The practical aspect of these calculations is to supply the data required for the preliminary design (which often becomes the final design).

The basis of the analysis is, first, to calculate the cross-flow velocity field around the isolated aircraft. The cross section of the aircraft is mapped into a slit by a sequence of conformal transformations. The method does not apply at a cross section containing part of the swept-back trailing edge of the wing. The cross-flow velocity field can then be constructed. In particular, this calculation can be carried out at points along the axis of the store. The store itself is then regarded as a slender body in the (nonuniform) flowfield of the aircraft. The distribution of loading along the axis of the store is then calculated by another application of slender body theory. Account is taken of the vortices trailing from the lifting surfaces on the store. The forces and moments are found by suitable integrations of the loading.

The theory in its present form postulates an inviscid flow around a slender body near $M = 1$. It is likely that both of these restrictions can be relaxed.

The calculations were programmed in FORTRAN for operation on the IBM 7094 computer. The sample problem was the configuration shown in Fig. 1. The cross section of the aircraft was defined by 24 points, and the store was defined by 11 stations. Each position of the store required about 5 min computer time. Most of the calculations are concerned with data describing the flowfield in the cross-section plane; hence, the flowfield at a grid of points in the cross section can be obtained with little more effort than for

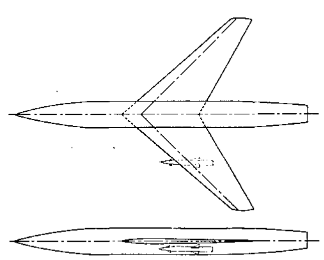

Fig. 1 Configuration of the aircraft-store combination.

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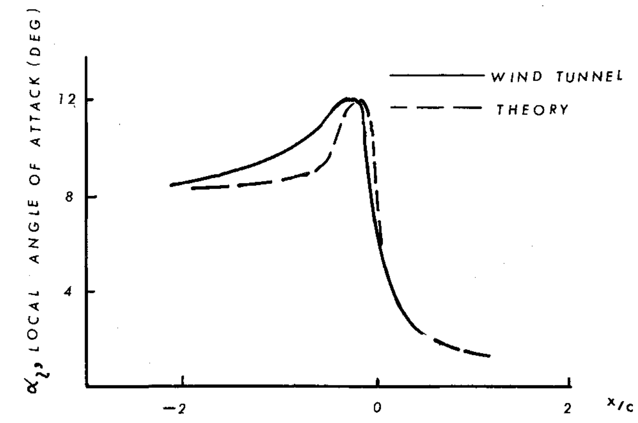

Fig. 2 Flowfield characteristics.

one point. Correspondingly, the aerodynamic forces on a store positioned at a grid of points in the cross section may require possibly 10–15 min computer time.

The program was applied to the configuration shown in Fig. 1 with the following data. The aircraft was at angle of attack 8°, sideslip angle 0°, the store axis was parallel to the fuselage reference axis and situated at 50% semispan on the left wing. If c denotes the wing chord at that station, then the axis of the store was below the wing at distance $0.145c$. The store axis was positioned at several points along this line characterized by

$$\frac{x}{c} = \frac{\text{distance from leading edge of } c \text{ to store c.g. (positive aft)}}{\text{chord at 50\% semispan}}$$

A comparison between the experimental[2] and theoretical values of the local angle of attack α_l at points along the axis of the store is shown in Fig. 2. The theory agrees with test in the maximum value of α_l and the place where the maximum occurs. There are differences in the gradient of α_l. We shall return to this point again in the text.

A comparison of the cross force is most conveniently made by combining the normal force coefficient C_N with the side force coefficient C_Y into a two-dimensional vector (C_Y, C_N) as shown in Fig. 3. Only the low-speed data[3] is presented be-

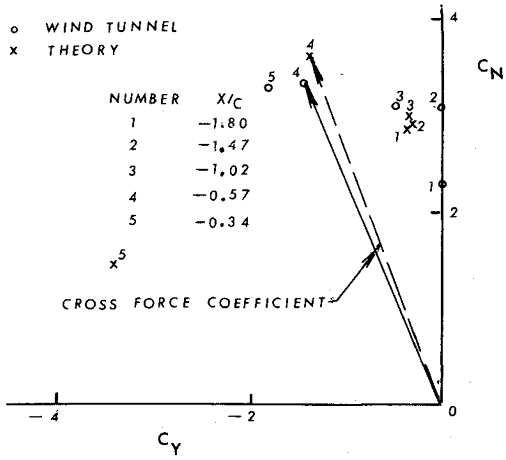

Fig. 3 Cross-force on the store.

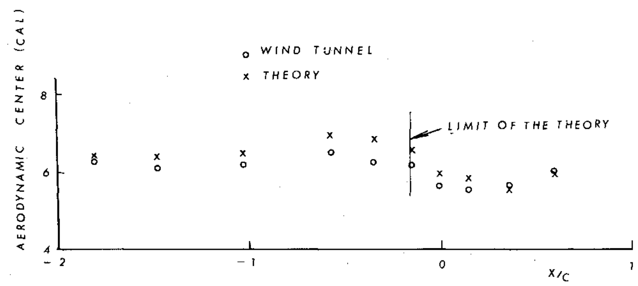

Fig. 4 Aerodynamic center of the store.

cause compressibility effects are minor[4]—a partial justification of slender body theory. A comparison of the experimental and theoretical vectors (as illustrated) suggests that the theory is sufficiently accurate for preliminary design purposes.

The pitching moment coefficient C_m and yawing moment coefficient C_n are similarly combined into a single vector $(C_m, -C_n)$. Define the "aerodynamic center" as that moment reference point on the axis of the store for which the resultant vectorial moment has a zero component along the cross-force vector. With this definition, the position of the aerodynamic center from the nose of the store is shown plotted in Fig. 4 for different (longitudinal) store locations.

The calculations are logically acceptable only when the trailing edge of the store is forward of the trailing edge of the wing ($x/c \leq 0.15$). However, if the calculations are continued aft of this point, the agreement appears quite satisfactory. This is so despite the disparity (Fig. 2) in the local flowfield. It is possible that the integrated effect of flow angularity is measured more adequately by the aerodynamic center of a store serving as a probe than by a rake.

References

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[2] Alford, W. J. and King, T. J., "Experimental Investigation of Flow Fields at Zero Sideslip, Near Swept- and Unswept-Wing-Fuselage Combinations at Low Speed," RM L56J19, Jan. 1957, NACA.

[3] Alford, W. J., "Experimental Static and Aerodynamic Forces During Simulated Launching from the Midsemispan and Wing-Tip Locations of a 45° Sweptback Wing-Fuselage Combination," RM L55A12, April 1955, NACA.

[4] Alford, W. J. and King, T. J., "Experimental Static Aerodynamic Forces and Moments at High Subsonic Speeds on a Missile Model During Simulated Launching from the Midsemispan Location of a 45° Sweptback Wing-Fuselage-Pylon Combination," RM L56J05, Jan. 1957, NACA.

Toward Simpler Prediction of Transonic Airfoil Lift, Drag, and Moment

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DESIGN of wings for operation at transonic speeds has been hampered by the inability to predict their aerodynamic characteristics accurately. The difficulties stem, as is widely known, from the inherent nonlinearity present in inviscid treatments,[1] the significant influence of viscosity in the flow outside the boundary layer, and coupling between the boundary-layer flow and the external flow. Attacks on the problem have usually followed one of two courses: Solve the inviscid problem by various numerical techniques[2] or employ semiempirical correlations[3] of experimental data.

Because experiments carried out in our laboratory convinced us that viscosity has effects on the pressure distribution even at low angles of attack that could not be ignored and because the results obtained with many entirely analytical methods are not as accurate as required for satisfactory design work and are, in addition, expensive to use, we chose the second course. Following this approach one has, of course, almost an infinite variety of data correlations and theoretical calculations to choose from in assembling a method to predict aerodynamic characteristics of an airfoil between $M = 0$ and $M = 1.0$. To test the reliability of the approach selected, only simple versions of its analytical portions were used. The results have been sufficiently promising that more accurate versions are now being incorporated into the method. The method consists of five steps:

1) Compute the $M = 0$ pressure distribution using a distribution of vortices along the chord.[4] Use the Karman-Tsien method to correct for changes in Mach number up to the critical. A more accurate version of this procedure[5] includes boundary layer displacement effects.

2) Compute the $M = 1.0$ pressure distribution using the empirical technique of Thompson and Wilby[2] or the analytical technique of Truitt[6] if $\alpha = 0$.

3) Compute the pressure distribution rearward from the airfoil crest for $M_{CR} \leq M \leq M = 1$ using the semi-empirical technique of Sinnott and Osborne.[7] Fitzhugh[3] also makes use of this technique. In the region between the sonic line and the shock it is reasonable to expect that ultimately a completely analytical technique could be developed to account for the wave interactions and thus to duplicate the results of Sinnott and Osborne.

4) Assume the pressure variation over the first 5% of the airfoil is constant for $M \geq M_{CR}$. Then use a cubic spline to represent the pressure distribution between $x/c = 0.05$ and the crest. The fit should match the value and slope of the pressure distribution at the airfoil crest as well as the pressure value at $x/c = 0.05$.

5) Compute the Reynolds Number at the shock. For laminar boundary layers assume the pressure rise through the shock begins 50δ upstream of the predicted shock location. The pressure then rises linearly to the value downstream of the shock at the predicted shock location.

$$\delta = 5.73x/(Re_x)^{1/2}$$

For turbulent boundary layers the pressure rise extends linearly over 5δ turbulent. Obviously, this is a very crude approximation. More accurate versions are given in the papers by Murphy and by Rose.[8]

The procedure represented by the five steps was programmed for solution on a digital computer. Complete distribution could be obtained for 10 Mach numbers at one angle of attack in 25 seconds on an IBM 370/165. The

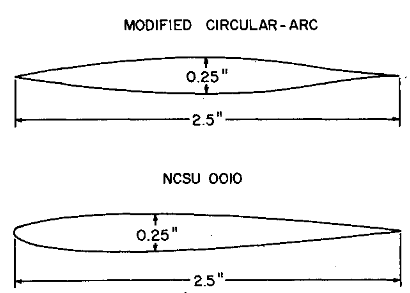

Fig. 1 Airfoil sections tested.

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