Aerodynamics of a Slender Cone with Asymmetric Nose Bluntness at Mach 14

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THE effect of nose shape asymmetries on the aerodynamics of a 10° cone ($d = 5$ in) was studied in ARL's 20-in.-hypersonic wind tunnel ($M_\infty = 14.2$, $p_t = 1500$ psi, $T_t = 2000$°R, $Re_{\infty,d} = 2.3 \times 10^5$, $T_w/T_t = 0.3$, $\gamma = 1.4$). Surface pressures were measured at various angles of attack and sideslip. Schlieren pictures were taken with a special nose model at several roll attitudes. Stability derivatives were measured by the free oscillation technique at the reduced frequency $\omega d/2V_\infty \approx 0.003$. The results are summarized and some conclusions relating to flight dynamics are drawn.

Figure 1 shows a sketch of model A and the pressure distribution over the afterbody. The asymmetry of the nose tip is the result of a planar cut 40° off the x-z plane. The projected frontal area of the cut is approximately $\frac{1}{2}$% of the base area. The schlieren picture on the left-hand side of Fig. 2 shows the shock contour in the x-y plane near the nose of model A. The plot below is a cross section of model and shock at the axial station $l/L = 0.2$. In the neighborhood of $\phi = 270°$, the observed shock nearly coincides with the sharp cone shock, as given by the expression[1]

$$r_{sh}/r = (3\gamma + 1)/(2\gamma + 2) + 1/2M_\infty^2 \tan^2\theta_c \qquad (1)$$

Correspondingly, the pressure over this ray is nearly constant. The two-dimensional character of the shock over the nose bevel proper explains the large shock radius downstream of the bevel and the blunt cone type of pressure distribution along the ray $\phi = 90°$, which shows a surprisingly large overexpansion. Circumferentially the effect of the nose asymmetry extends beyond the top and bottom rays. The lateral pressure differences $p_{90} - p_{270}$ associated with the asymmetry of the shock contour produce a large sideforce and side moment. The latter referenced to a moment center at $l/L = 0.65$ may exceed the moment resulting from the force acting on the nose bevel, and, in free flight, the vehicle would trim to a positive β. Such a trim direction has been observed[2] for a similar configuration. The pressure distribution over the ray $\phi = 90°$ is independent of α up to about $\pm$ 4°, but changes at larger angles of attack so that the side force is expected to reduce in magnitude or may even reverse its direction, while the negative side moment increases. The pressure differences between windward and leeward rays are symmetric about $\alpha = 0$, when $\beta = 0$; but they are asymmetric functions of the angle of sideslip, when $\alpha = 0$. Also, the pressure distributions along $\phi = 0$ and 180° change from blunt cone character ($\beta \leqq 0$) to sharp cone character ($\beta > 0$).

Considering the asymmetric pressure distribution, it was expected that the stability derivatives in pitch and yaw would be unequal in magnitude for axial flow condition and that C_{m_α} and C_{n_β} would vary unsymmetrically with β over the range of positive and negative angles of sideslip, but symmetrically with respect to α. The static derivatives obtained from the dynamic tests, shown in Fig. 3, confirm this hypothesis. The most striking result is the difference of 30% between C_{m_α} and C_{n_β} for $\alpha = \beta = 0$.

Referring now to model B, Fig. 4, the right-hand portion of the nose is the result of a planar cut perpendicular to the x axis, while the left hand portion of the nose again results from a planar cut 40° off the x-z plane. The nose portion extending in

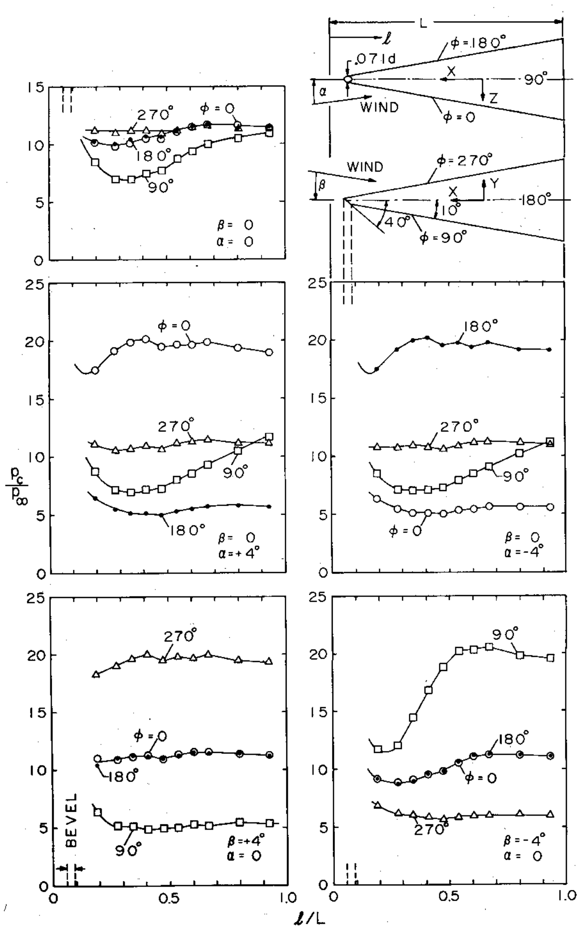

Fig. 1 Model A and pressure distributions at various angles of attack and sideslip.

the $+y$ direction therefore makes a greater contribution to the nose drag then the other side. Nevertheless, the shock radius at the station $l/L = 0.2$ is larger downstream of the bevel, Fig. 2. For the $+y$ direction, the observed shock contour is well described by the following equation holding for axisymmetric bluntness[3],

$$r_{sh}/d_N = 1.0C_D^{\ 1/4}(x/d_N)^{0.43} \qquad (2)$$

($C_D = 1.7$, $d_N = 0.07d$, $x =$ axial distance from shock apex).

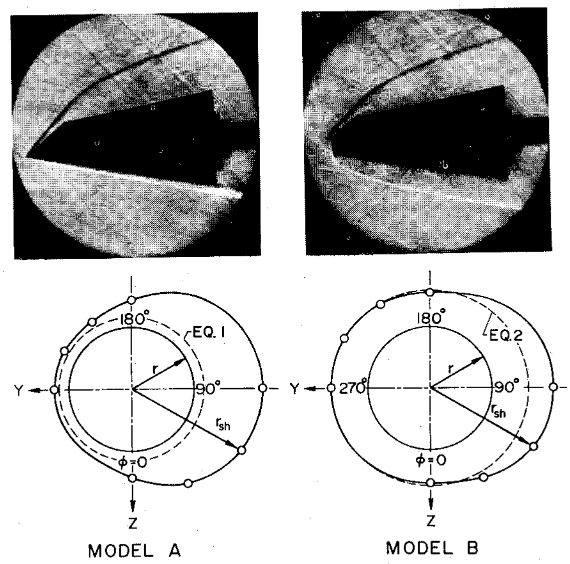

Fig. 2 Selected schlieren pictures and cross sections of model and bow shock at the axial station $l/L = 0.2$; $\alpha = \beta = 0$.

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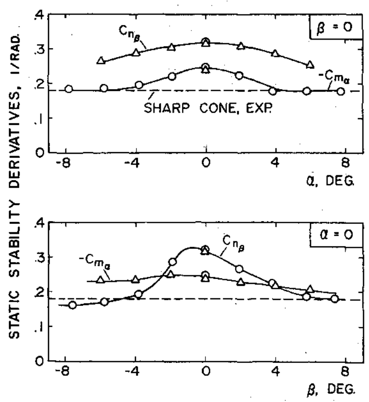

Fig. 3 Static stability derivatives of Model A, (moment coefficients are referenced to base area and base diameter, moment center at $l/L = 0.65$).

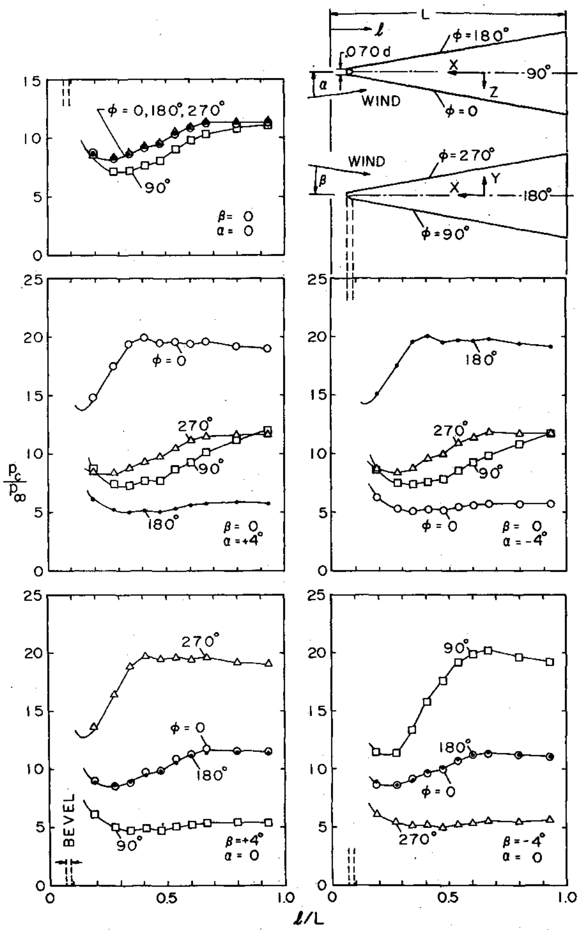

Fig. 4 Model B and pressure distributions at various angles of attack and sideslip.

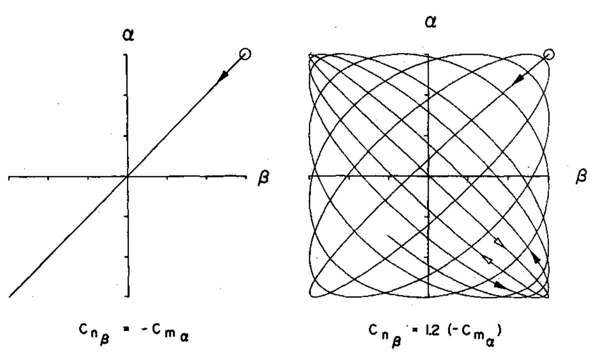

Fig. 5 Effect of equal and nonequal derivatives in pitch and yaw on the motion in the $\alpha - \beta$ plane, ($I_y = I_z$, $p = 0$, $\alpha_0 = \beta_0$, $\dot{\alpha}_0 = \dot{\beta}_0 = 0$).

Equation (2), however, does not apply for the $-y$ side, where the shock radius is again governed by the planar slanted nose cut. An asymmetric surface pressure distribution results from the rotational asymmetry of the shock. For $\alpha = \beta = 0$ the largest overexpansion is observed along the ray $\phi = 90°$, which is downstream of the low drag portion of the nose. The centroid of the resulting side load distribution may coincide with the center of gravity, in which case the vehicle would not trim. If mass asymmetry is also present, the side force combined with the c.g. offset along the z axis will provide a roll torque that would spin a rolling vehicle up or down. This example shows, that the unfavorable condition of "roll through zero" can occur without trim. The rotational symmetry of the stability derivatives is again destroyed by the small configurational asymmetry of the nose; $C_{n_\beta} = 1.17(-C_{m_\alpha})$ was measured for $\alpha = \beta = 0$.

The effect of unequal values of C_m and C_{n_β} on the angular motion is illustrated in Fig. 5 by the very simple case of a nonrolling vehicle with $I_y = I_z$. Damping is neglected and zero trim is assumed. The initial conditions are $\alpha_0 = \beta_0$, $\dot{\alpha}_0 = \dot{\beta}_0 = 0$. When $C_{n_\beta} = -C_{m_\alpha}$, constant over a small range of angles, the motion pattern in the $\alpha - \beta$ plane is a straight line, as predicted by the tricyclic theory,[4] and this pattern is repeated with every cycle of oscillation. When $C_{n_\beta} \neq -C_{m_\alpha}$, the motion pattern is a Lissajous figure that is not predictable by the tricyclic theory because its basic assumption of aerodynamic rotational symmetry does not hold. For $C_{n_\beta} = 1.2(-C_{m_\alpha})$, the period of a yaw cycle is 10% shorter than the period of a pitch cycle, and the motion character changes continuously. Sharp loops are followed by rounder loops and round ones become sharp again. Upon completion of this series of loop changes, the angular motion reverses its direction.

Physically, the investigated models are properly described as bodies of revolution having a small configuration asymmetry. Aerodynamically, however, they are not bodies of revolution because the configuration asymmetry is at the nose.

References

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Instability of Hypersonic Viscous Shock Layer with Finite Rate Chemistry

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Nomenclature

Re = shock Reynolds number per foot of body radius = $\rho_\infty U_\infty / \mu_1$
ρ_∞ = density of the gas in the freestream
U_∞ = flight speed
μ_1 = viscosity of the gas immediately behind the shock front
T = temperature

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