[3] Demetriades, A., "An Experimental Investigation of the Stability of the Hypersonic Laminar Boundary Layer," Memo 43, 1958, Guggenheim Aeronautical Lab., California Institute of Technology.

[4] Potter, J. L. and Whitfield, J. D., "Effects of Unit Reynolds Number, Nose Bluntness, and Roughness on Boundary Layer Transition," TR-60-5, 1960, Arnold Engineering Development Center.

[5] Nagamatsu, H. T., Graber, B. C., and Sheer, R. E., Jr., "Critical Layer Concept Relative to Hypersonic Boundary Layer Stability," Rept. 66-C-192, 1966, General Electric Co.

[6] Maddalon, D. V. and Henderson, A., Jr., "Boundary-Layer Transition on Sharp Cones at Hypersonic Mach Numbers," *AIAA Journal*, Vol. 6, No. 3, March 1968, pp. 424–431.

[7] Softley, E. J. and Graber, B. C., "Experimental Observation of Transition of the Hypersonic Boundary Layer," *AIAA Journal*, Vol. 7, No. 2, Feb. 1969, pp. 257–263.

[8] Deem, R. E. and Murphy, J. S., "Flat Plate Boundary Layer Transition at Hypersonic Speeds," AIAA Paper 65-128, New York, 1965.

[9] Sanator, R. J., DeCarlo, J. P., and Torrillo, D. T., "Hypersonic Boundary-Layer Transition Data for a Cold-Wall Slender Cone," *AIAA Journal*, Vol. 3, No. 4, April 1965, pp. 758–760.

[10] Rouse, H., "A General Stability Index for Flow Near Plane Boundaries," *Journal of the Aeronautical Sciences*, Vol. 12, No. 4, Oct. 1945, pp. 429–431.

[11] Rouse, H., *Elementary Mechanics of Fluids*, Wiley, New York, 1946, p. 172.

[12] Cohen, N. B., "Boundary Layer Similar Solutions and Correlation Equations for Laminar Heat-Transfer Distributions in Equilibrium Air at Velocities up to 41,000 Feet Per Second," TR R-118, 1961, NASA.

[13] van Driest, D. R. and Blumer, C. B., "Boundary Layer Transition: Freestream Turbulence and Pressure Gradient Effects," *AIAA Journal*, Vol. 1, No. 6, June 1963, pp. 1303–1306.

[14] Iverson, H. W. and Hsu, T.-T., "A Local Reynolds Number to Predict Transition from Laminar to Turbulent Flow," *Heat Transfer Thermodynamics and Education*, McGraw-Hill, New York, pp. 207–212.

Real Gas Effects on Shock-Tube Flow Nonuniformity

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AN understanding of flow nonuniformity is necessary for meaningful shock-tube testing. One important source of flow nonuniformity is the boundary-layer build-up near the shock-tube wall. The experiments of Duff[1] showed that the shock and contact surfaces approach a limiting separation distance and that the density increases significantly between shock and contact surfaces. This behavior was described analytically by the work of Roshko[2] and Mirels.[3,4] Reference 4 presents calculated results based on an ideal gas for a range of shock Mach numbers. Bertin[5] analyzed boundary-layer induced property variations in a circular shock tube for equilibrium real air operation at shock Mach numbers up to 9.5. Both papers indicate a definite decrease in flow nonuniformity with increase in Mach number.

The present work was undertaken to determine the effect of real gas behavior on flow nonuniformity over a range of Mach numbers. Real gas effects increase with increasing shock Mach number. Physical considerations indicate that the phenomena of dissociation and vibrational excitation would tend to lessen flow nonuniformity. To predict the magnitude of this lessening as a function of shock Mach numbers, the method of Ref. 4 was extended to include real gas effects. Equilibrium flow was considered over a range of Mach numbers. The concept of an equivalent inviscid channel (as described in Ref. 4 and illustrated in Fig. 1) was used for all cases. With the assumption of laminar flow, the channel area can be described by the equation

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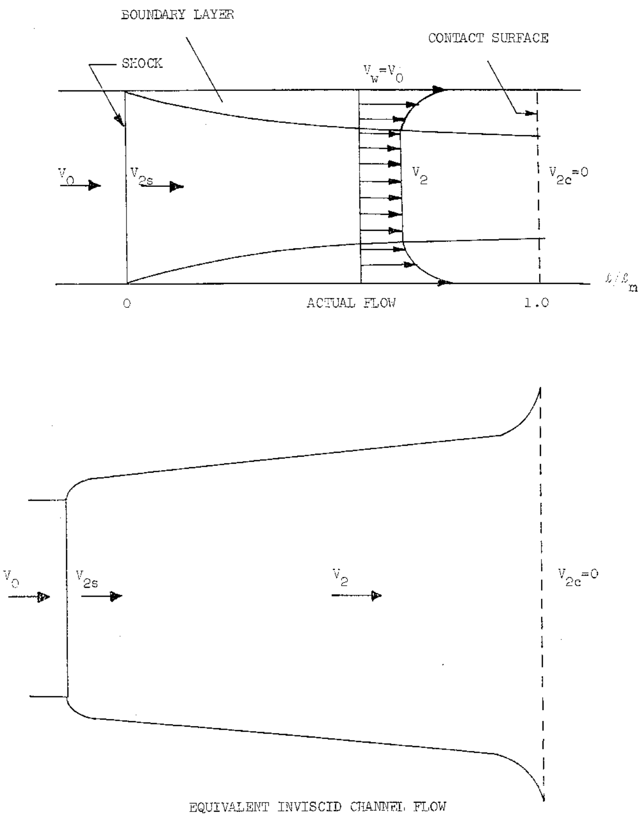

Fig. 1 Actual and equivalent inviscid channel representation of flow in fixed shock coordinates.

$$A_{2s}/A_2 = 1 - (l/l_m)^{1/2} \tag{1}$$

where A_{2s} = cross-sectional area of the channel at the shock front, A_2 = cross-sectional area at distance l behind the shock, l = distance behind the shock, l_m = maximum separation distance between the shock and the contact surface. Calculated results based on such an inviscid channel and presented in Ref. 4 give the variation in flow properties between the shock and contact surface for a laminar boundary layer in an ideal gas. These results are compared to the real gas calculations of the present investigation.

The "ideal dissociating gas" model of Lighthill[6] was used to represent nitrogen in chemical equilibrium for conditions at which dissociation occurs. The accurate range of approximation using this model has a lower limit due to its assumptions regarding vibrational energy and an upper limit imposed by the neglecting of excited electronic states. To obtain reasonably accurate representation using this model, calculations were restricted to shock Mach numbers from 10 to 20.

The assumption of chemical equilibrium is valid provided the length of the relaxation zone λ behind the normal shock is small compared to the maximum separation distance between the shock and the contact surface. Experimental values for the relaxation length in shock-heated nitrogen were found by Allen, Keck, and Camm[7] for initial pressures of 1, 3, and 10 mm of mercury and shock Mach numbers in the range considered in the present work. The values of l_m can be obtained using the results of Ref. 3. Comparison of the values of λ and l_m for an initial pressure of one millimeter of mercury show that for Mach number of 20 in a 4-in.-diam shock tube, the relaxation zone length is less than one percent of l_m. For a Mach number of 15 the relaxation zone occupies

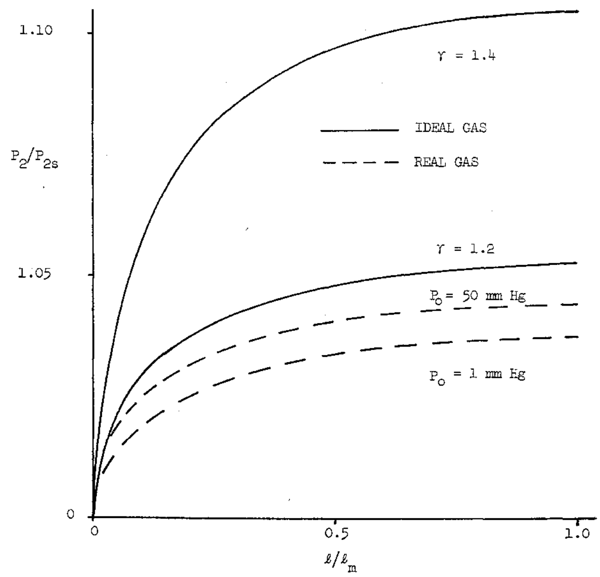

Fig. 2 Pressure variation between shock and contact surface for a shock Mach number of 20 in nitrogen.

approximately 6% of the test region. The influence of the relaxation zone increases as the initial pressure or shock Mach number decreases or the shock-tube diameter decreases. For the Mach number and pressure range considered in this paper, the equilibrium assumption appears valid except for the possible exception of the low-pressure, low Mach number range near $P_0 = 1$ mm Hg, $M = 10$.

A shock velocity and initial conditions of the undisturbed gas were assumed to start each calculation. Conditions immediately behind the shock were found by using an iterative scheme involving the conservation equations and the property relations of an "ideal dissociating gas." Composition and thermodynamic properties of the ideal dissociating gas are related by the following expressions:

$$y^2/(1 - y) = (\rho_D/\rho) \exp(T/\tau_D),\ P = \rho RT(1 + y)$$
$$h = (4 + y)RT + U_D y \tag{2}$$

where y = mass fraction of atomic nitrogen and ρ_D, T_D, U_D =

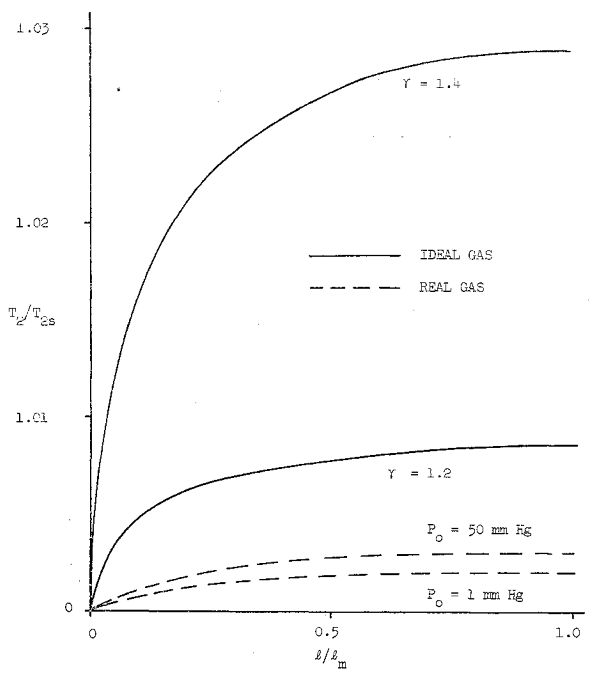

Fig. 3 Temperature variation between shock and contact surface for a shock Mach number of 20 in nitrogen.

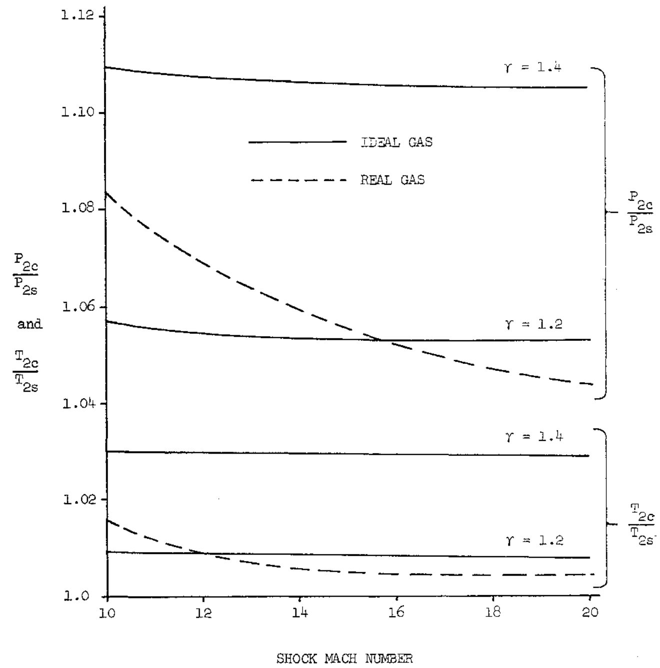

Fig. 4 Maximum variation in temperature and pressure between shock and contact surface as a function of Mach number.

characteristic parameters of an ideal dissociating gas. Flow in the region between the shock and the contact surface was calculated using the continuity equation

$$\rho_{2s} v_{2s} A_{2s} = \rho_2 v_2 A_2 \tag{3}$$

the energy equation,

$$h_{2s} + \tfrac{1}{2} v_{2s}^2 = h_2 + \tfrac{1}{2} v_2^2 \tag{4}$$

the area relationship [Eq. (1)] and the condition of constant entropy along the channel. This last condition required the use of the expression for entropy of an ideal dissociating gas which takes the form

$$S = 3 \log T + y(1 - 2 \log y) - (1 - y) \log(1 - y) - (1 + y) \log\rho + \text{const} \tag{5}$$

Solutions were obtained at regular intervals between the shock and contact surface for a range of shock Mach numbers from 10 to 20 and initial pressures of 1 and 50 mm of mercury. Nonuniformity in temperature and pressure predicted by the real gas model is compared to the ideal gas results for a shock Mach number of 20 in Figs. 2 and 3. For this Mach number it is apparent that the calculated flow nonuniformity is substantially reduced by inclusion of dissociation effects. It is also of interest to determine the variation of flow nonuniformity with Mach numbers for the real gas model. This behavior is displayed and compared to the ideal gas results in Fig. 4. For the real gas the flow nonuniformity continues to decrease as Mach number increases. The ideal gas calculations predicted very little change in degree of nonuniformity between shock Mach numbers of 10 and 20. The calculated results indicate, as might be expected, that as the shock Mach number increases to high values, the process of dissociation acts as an energy sink to lower the gas temperature and lessen the flow nonuniformity.

References

[1] Duff, R. E., "Shock Tube Performance at Low Initial Pressure," *The Physics of Fluids*, Vol. 2, No. 2, March-April 1959, pp. 207–216.

[2] Roshko, A., "On Flow Duration in Low Pressure Shock

Tubes," *The Physics of Fluids*, Vol. 3, No. 6, Nov.-Dec. 1960, pp. 835–842.

[3] Mirels, H., "Test Time in Low Pressure Shock Tubes," *The Physics of Fluids*, Vol. 6, No. 9, Sept. 1963, pp. 1201–1214.

[4] Mirels, H., "Flow Nonuniformity in Shock Tubes Operating at Maximum Test Times," *The Physics of Fluids*, Vol. 9, No. 10, Oct. 1966, pp. 1907–1912.

[5] Bertin, J. J., "Analysis of Test Times and Boundary Layer Induced Property Variations in a Circular Shock Tube," TN-D-3759, Jan. 1967, NASA.

[6] Lighthill, M. J., "Dynamics of a Dissociating Gas, Part I Equilibrium Flow," *Journal of Fluid Mechanics*, Vol. 2, No. 1, Jan. 1957, pp. 1–32.

[7] Allen, R. A., Keck, J. C., and Camm, J. C., "Nonequilibrium Radiation and the Recombination Rate of Shock-Heated Nitrogen," *The Physics of Fluids*, Vol. 5, No. 3, March 1962, pp. 284–291.

Effect of Flow Discontinuities on Oscillatory Pressures Measured in Supersonic Wind Tunnels

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In 1959, Mollø-Christensen and Martuccelli[1] pointed out that surface-pressure measurements on an oscillating model in a supersonic wind tunnel may be greatly affected by the presence of possible wind-tunnel flow discontinuities. They further suggested that this fact may be responsible for the lack of any published experimental data on supersonic pressure distributions on oscillating wings. This situation still exists today. Therefore, it may be appropriate to present some new evidence, which indicates that the effect of flow discontinuities does not always have to be very large, and which suggests a practical method for assessing the magnitude of the effect and to determine suitable corrections if any are required.

Pressure interference associated with tunnel-flow discontinuities usually is very small as compared to steady surface pressures, but may become significant when compared to the much smaller oscillatory surface pressures. This interference effect is caused primarily by the fact that, in general, a point on the surface of an oscillating model continuously changes its location in the wind tunnel. Thus, the measured surface pressure at that point is a function of both the instantaneous mean flow conditions as given by the attitude and motion of the model and of any local flow discontinuities which may be present along the path of the point of measurement. To separate these two effects, an experimental condition must be found where only one of them is present or else where they are both present but in a known combination. The method described here requires a separate experiment during which the model performs additional oscillation in such a way that the mean flow conditions remain practically constant while the point of measurement traverses the same flow discontinuities as those intersected during the primary experiment. In this way, the aerodynamic effect is rendered negligible and the additional experiment gives only the interference effect to be determined.

Table 1 Experimental and theoretical values of oscillatory pressure ratio P_{osc}/P_0

	α_{mean}	P_{osc}/P_0 Pitching oscillation $\Delta\theta = 1.5°$	P_{osc}/P_0 Longitudinal oscillation $\Delta x = 0.019\,c$
Experiment, sidewall	0.14°	0.0056	0.0003
Experiment, topwall	0.30°	0.0058	0.0001
Experiment, topwall	−0.94°	0.0053	...
Linear theory	0	0.010	...
Nonlinear theory	0	0.005	...

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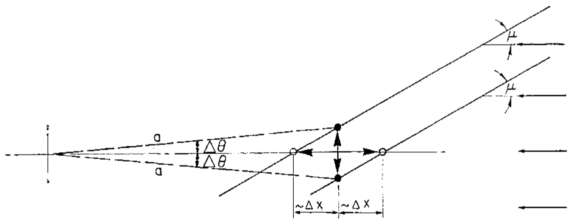

Fig. 1 Determination of equivalent amplitude of longitudinal oscillation; refraction across bow shock wave neglected.

If the primary experiment is an oscillation-in-pitch, the additional experiment may be a translational oscillation in the longitudinal direction‡ of the model, with suitably adjusted amplitude. If the flow discontinuities are assumed to be weak enough to propagate as Mach waves (a reasonable assumption in a good wind tunnel) and if the refraction of the flow discontinuities through the bow shock wave of the model is neglected, the required amplitude Δx of the longitudinal oscillation may be obtained from

$$\Delta x = a(\Delta\theta)\cot\mu \qquad (\Delta\theta \ll 1) \tag{1}$$

where $\mu = \sin^{-1}(1/M)$ is Mach angle, a is the distance of the point of measurement from the axis of oscillation and $\Delta\theta$ is the amplitude of the pitching oscillation (see Fig. 1).

The effect on Δx of refraction through a moving bow shock wave can be shown to be small for relatively weak bow shocks (e.g., such as on wings with sharp leading edge and relatively small nose angle) and small amplitudes. For instance, in the present experiments, this effect was of the order of 5% and could be neglected. In cases where the foregoing conditions are no longer met, a more detailed calculation of Δx may be required with refraction taken into account. This may be particularly important when the effects of flow discontinuities are large and irregular and when the point of measurement is far downstream from the leading edge.

Oscillatory pressure measurements were performed on a half model of an aspect ratio 3 cropped delta wing with taper ratio 0.072 and 5% thick biconvex profile in the Mach 1.8 nozzle of the NAE 30-in. × 16-in. suction wind tunnel. The model was mounted either on the flat sidewall opposite to the single nozzle wall of the tunnel or on the topwall of the tunnel. The model was oscillated in pitch with an amplitude of ±1.5° around an axis 0.45 c behind the wing apex, where c is the wing root chord. In addition, it was also oscillated (at slightly different frequency) in longitudinal translation, with the amplitude determined by means of Eq. (1). The quantities measured are shown in Fig. 2, where trace 1 is model deflection in pitch, traces 2 and 3 are oscillatory surface pressures in pitching and longitudinal oscillations, respectively, and trace 4 is aerodynamic and electronic noise measured

‡ At hypersonic speeds, it may be better to perform the additional experiment as a slow plunging oscillation, since otherwise the required amplitude could become impractically large.