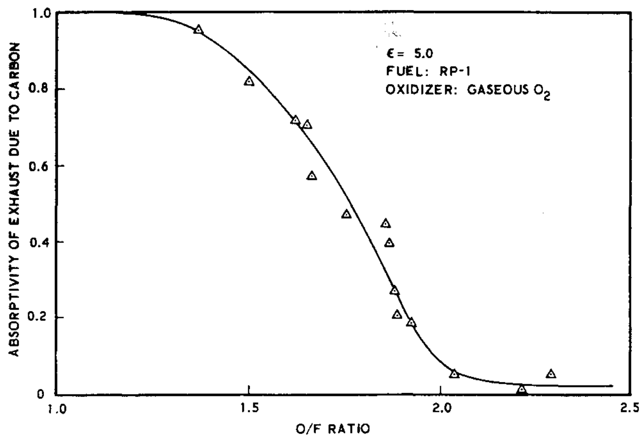

Fig. 4 Absorptivity due to solid carbon in the RP-1gaseous oxygen exhaust as a function of oxidizer-to-fuel ratio.

The transmissivity measurements were obtained through open ports located at area ratio 5, where the nozzle pressure is close to atmospheric pressure for typical tests. The transmission at 5911 Å was used here, although the spectrometer also scanned other wavelengths. A schematic diagram of the experimental system has been presented previously.[9]

The particle size distribution used in these tests was the same as that used in other experiments[7,9]; the volume to surface mean diameter is about 3 μ. The experimental data (including some extinction data points reported previously[7]) are shown in Fig. 3 compared with theoretical lag and zero lag predictions based in Eqs. (10) and (11). Although there is appreciable scatter of the data, the lag theory seems to provide a reasonable correlation.

The experimental data of Fig. 3 are not the raw data; a correction was made for extinction caused by solid carbon in the flow. This was obtained by measuring the extinction due to solid carbon for a series of tests employing RP-1 and gaseous oxygen with no oxide particles present. The results of these tests are presented in Fig. 4. The correction is made assuming the solid carbon is distributed homogeneously across the plane of measurement and using the relationship

$$1 - \alpha' = (1 - \alpha_{MgO})(1 - \alpha_C) \qquad (12)$$

where α' is the total extinction due to both carbon and MgO. It was assumed that extinction due to carbon did not change as oxide particles were added to the flow, but was dependent only on the oxidizer-to-fuel ratio.

Summary

Light extinction measurements performed in the nozzle of a rocket exhaust containing metallic oxide particles of a known size distribution have agreed with theoretical predictions based upon velocity lag theory. This establishes further evidence that such theories provide reasonable descriptions of the actual gas-particle flow processes occurring in two-phase rocket nozzle flows.

References

[1] Jones, W. H., "Solid propellant exhaust simulation," AIAA J. **1**, 721–722 (1963).

[2] Hoglund, R. F., "Recent advances in gas-particle nozzle flows," ARS J. **32**, 662–671 (1962).

[3] Carlson, D. J. and Hoglund, R. F., "Sphere drag and heat transfer relationships for gas-particle nozzle flows," AIAA J. **2**, 1980–1984 (1964).

[4] van de Hulst, H. C., *Light Scattering by Small Particles* (John Wiley and Sons, Inc., New York, 1957), Chap. 9, pp. 114–128.

[5] Gray, D. E. (ed.), *American Institute of Physics Handbook* (McGraw-Hill Book Co., Inc., New York, 1963), 2nd ed., Chap. 6, p. 29.

[6] Stull, V. R. and Plass, G. N., "Emissivity of dispersed carbon particles," J. Opt. Soc. Am. **50**, 121 (1960).

[7] Carlson, D. J., "Experimental determination of thermal lag in gas-particle nozzle flow," ARS J. **32**, 1107 (1962).

[8] Carlson, D. J., "A high-resolution rapid-scan spectrometer," Aeronutronic Publ. U-2002 (January 31, 1963).

[9] Carlson, D. J., "Emittance of condensed oxides in solid propellant combustion products," Tenth Symposium (International) on Combustion, Cambridge, England, Paper 119 (August 17–21, 1964).

Step Induced Boundary-Layer Separation Phenomena

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Nomenclature

h = step height
P = plate pressure
P_p = plate "plateau" pressure
P_∞ = freestream pressure
δ = boundary-layer height

A NUMBER of experimental[1–3] and theoretical[4–6] studies have been conducted on the separation phenomena associated with two-dimensional steps and planar shocks interacting with a turbulent boundary layer. Much less study has been centered upon the more complicated phenomena involving a three-dimensional protuberance in a supersonic flow. In order to provide information on the effects of circumferential flow on the separated region formed upstream of a protuberance, a wind tunnel† test program was conducted utilizing two-dimensional and cylindrical steps immersed in supersonic flow. The Mach number and turbulent boundary-layer height in the test section were measured and found to be 2.71 and 0.19 in., respectively. The step heights ranged from 0.1 in. (h/δ = 0.525) to 0.5 in. (h/δ = 2.63); the cylindrical step diameters ranged from 0.5 to 1.0 in. A sketch of the test models is shown in Fig. 1.

The results of the two-dimensional step tests were found to correlate well with those of previous investigations. The pressure rise at the step face established an oblique shock wave that propagated upstream and caused the boundary layer to separate from the base plate. The length of the high-pressure region downstream of the shock wave (hereafter known as the compression region) was found to be a function of the step

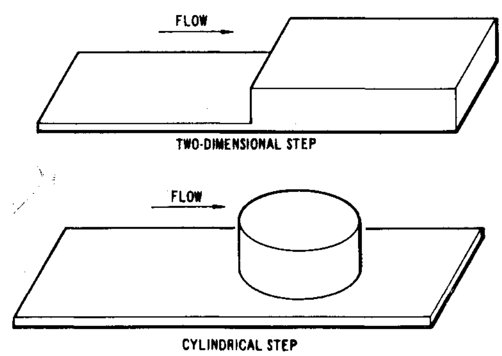

Fig. 1 Sketch of test models.

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† The work was performed in the University of California at Los Angles supersonic wind tunnel.

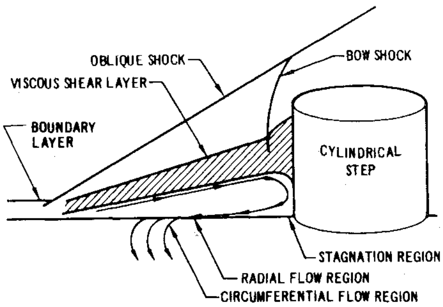

Fig. 2 Flow model associated with cylindrical steps in a supersonic airstream.

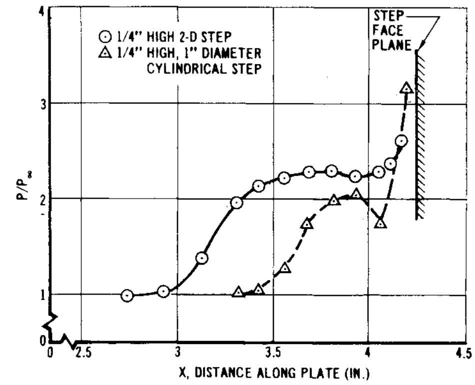

Fig. 3 Comparison of plate pressure profiles for two-dimensional and cylindrical steps.

to boundary-layer height ratio, whereas the "plateau" pressure reached in this region was primarily determined by the upstream Mach number.

Circumferential flow and lateral pressure relief are, of course, the difference between the separation phenomena associated with the presence of cylindrical as opposed to two-dimensional steps. It was found that three distinct flow regions exist between the oblique shocks and cylindrical steps. The outermost region is located immediately behind the shock wave that is formed upstream of the step and is made up of flow directed circumferentially around the cylinder. It is in this region that the plate pressure builds up to a first peak. The second region, inside the first, is characterized by radial flow directed away from the step along the plate. It is felt that this is a rather high-speed vortex region in which a sizable dip in plate pressure occurs. A third region of presumably low-speed, high-pressure flow is apparent at the step-plate junction. The same type of pressure profile has been noted by Burbank et al.[7]

The three region hypothesis was developed as a result of the examination of pressure data, schlieren photographs, and oil flow studies. Table 1 presents the results of this examination in tabular form. A proposed flow model incorporating all available data is shown in Fig. 2. Note that the boundary layer separates from the plate and becomes a viscous shear layer that impinges upon the front face of the cylindrical step and induces the formation of a normal shock wave. This shock interacts with the primary oblique shock wave to produce a lambda shock pattern. Conversely, schlieren photographs indicate that the viscous shear layer is forced over the top of the two-dimensional steps, and only the low-energy separated flow impinges against the step face. It has therefore been theorized that the pressure relief offered by circumferential flow around the sides of a cylinder permits the compression region to shrink in size and allows the high-speed viscous shear layer to impinge upon the step face. This establishes a second shock and causes a second high plate pressure region to form immediately upstream of the step. A comparison of the plate pressures associated with two-dimensional and cylindrical steps is shown in Fig. 3. It can be seen that the "plateau" pressures are higher for the two-dimensional step, but the plate pressures at the step-wall junction are much higher for the cylindrical protuberance. It is also obvious that in reducing the cylinder diameter from infinity (two-dimensional) to a small multiple of the boundary-layer thickness, the undisturbed flow is allowed to approach the step face much more closely before being subject to a strong compression.

In conclusion, it must be added that the phenomena described may be a function of model size. For example, if the two-dimensional steps considered were very high in comparison to the boundary layer, the viscous shear layer may be forced to impinge against the step face. This would establish a lambda pattern similar in nature to that produced by a cylindrical step. Because of size limitations in the wind tunnel utilized, this possibility could not be investigated and remains a subject for further study.

Table 1 Results from step tests

A Two-dimensional

Height, in.	h/δ	oblique shock angle[a]	P_p/P_∞	Length of compression region,[a] in.
0.10	0.525	32°	2.23	0.63
0.25	1.32	33°	2.30	1.13

B Cylindrical

Height, in.	h/δ	Step diam, in.	Oblique shock angle[a]	P_p/P_∞	Length of compression region,[a] in.	Length of radial flow region,[b] in.	Pressure at inception of radial flow, P/P_∞
0.10	0.525	1.0	30°	2.02	0.41	0.04	2.02
0.10	0.525	0.75	29°	1.98	0.41	...	...
0.10	0.525	0.50	28°	1.91	0.41	...	...
0.25	1.32	1.0	31°	2.06	0.78	0.32	2.05
0.25	1.32	0.75	30.5°	2.04	0.69	0.20	2.00
0.25	1.32	0.50	30°	1.96	0.59	...	...
0.50	2.63	1.0	31°	2.15	1.19	0.58	2.10
0.50	2.63	0.75	30.5°	2.07	1.03	...	...
0.50	2.63	0.50	30°	2.03	0.85	0.50	1.98

[a] Results based on schlieren photographs.
[b] Results based on oil flow photographs.

References

[1] Bogdonoff, S. M., Kepler, C. E., and Sanlorenzo, E., "A study of shock wave turbulent boundary layer interaction at $M = 3$," Princeton Univ., Dept. of Aeronautical Engineering, Rept. 222 (1953).

[2] Chapman, D. R., Kuehn, D. M., and Larson, H. K., "Investigation of separated flows in supersonic and subsonic streams with emphasis on the effect of transition," NACA Rept. 1356 (1958).

[3] Kepler, C. E. and Bogdonoff, S. M., "Interaction of a turbulent boundary layer with a step at $M = 3$," Princeton Univ., Dept. of Aeronautical Engineering, Rept. 238 (1953).

[4] Mager, A., "On the model of a free shock-separated turbulent boundary layer," J. Aeronaut. Sci. **23**, 181–184 (1956).

[5] Mager, A., "Prediction of shock-induced turbulent boundary layer separation," J. Aeronaut. Sci. **22**, 201–202 (1955).

[6] Crocco, L. and Probstein, R. F., "The peak pressure rise across an oblique shock emerging from a turbulent boundary layer over a plane surface," Princeton Univ., Dept. of Aeronautical Engineering, Rept. 254 (1954).

[7] Burbank, P. D., Newlander, R. A., and Collins, I. K., "Heat transfer and pressure measurements on a flat plate surface and heat transfer measurements on attached protuberances in a supersonic turbulent boundary layer at Mach numbers of 2.65, 3.51, and 4.44," NASA TND-1372 (1962).

Measurements of Panel Response to Turbulent Boundary-Layer Excitation

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Nomenclature

f	=	frequency in cycles per second
M	=	Mach number
$\overline{p_w^2}$	=	mean-square wall pressure fluctuation
PWL	=	total acoustic power radiated
R	=	correlation coefficient
U	=	duct centerline velocity
U_c	=	convection velocity
$\langle Y^2 \rangle$	=	mean-square panel displacement
$\langle \bar{Y}^2 \rangle$	=	total mean-square panel displacement
ξ	=	incremental distance in direction of flow
τ	=	time delay
τ_w	=	wall shear stress

SOME preliminary results of an experimental investigation are presented concerning the motion induced by a turbulent boundary layer in rigidly supported panels and the radiated acoustic field generated by such motion. The measurements were conducted at the Boeing Boundary Layer Test Facility consisting of 25 ft of duct with cross sectional area of approximately 7.3 × 3.5 in. with velocities up to 700 fps. The thicknesses of the panels tested were 0.020, 0.040, 0.060 and 0.080 in. They were made by milling out an area of 7 × 12 in. from $\frac{3}{4}$-in. aluminum plates to the required thickness.

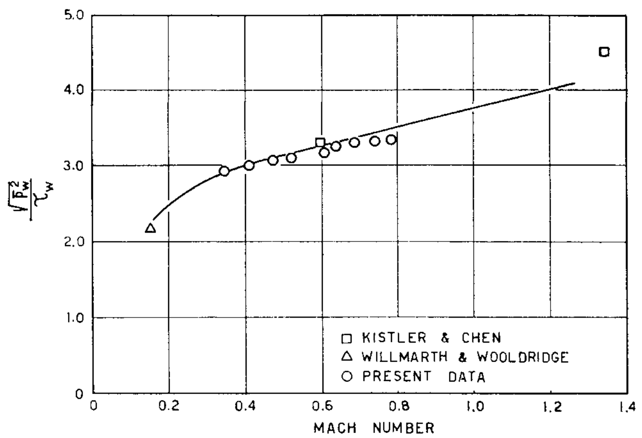

Fig. 1 Wall pressure fluctuations.

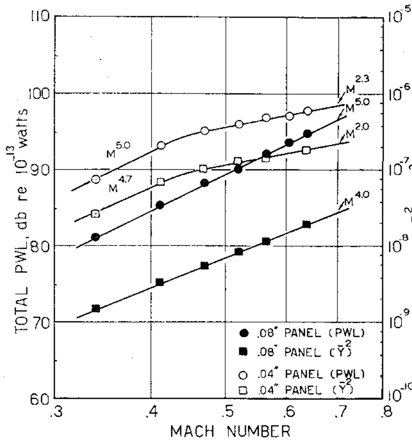

Fig. 2. Total acoustic power, PWL total mean square displacement, $\langle \bar{Y}^2 \rangle$.

Forcing Field

A careful study of the pertinent parameters of the turbulent field in the duct was made with the following results: 1) the mean velocity distribution and the shear stress τ_w obtained from the standard technique were found to be consistent with the published result of turbulent duct flows; 2) the measured mean square pressure fluctuation $\overline{p_w^2}$ on the rigid wall of the duct are in excellent agreement with those presented by various investigators[1, 2] (Fig. 1); 3) the space-time correla-

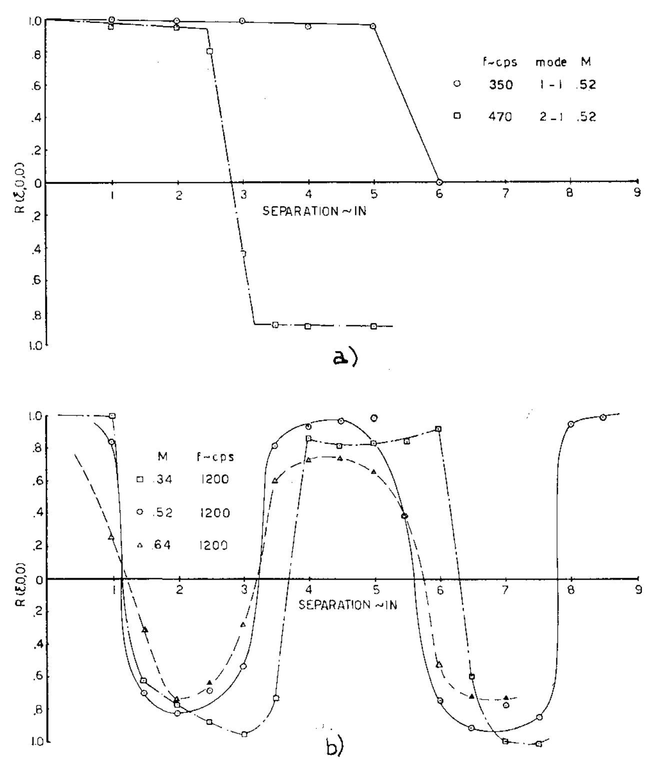

Fig. 3 Longitudinal space correlation of the displacement of a 0.040-in. panel: a) space correlation for Mach number 0.52 at various modal frequencies; b) space correlation for 1200 cps mode at various Mach numbers.

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