Rocket Exhaust Flow in Tube Launchers

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Theme

THE structural design of a lightweight, cylindrical-tube rocket launcher requires the calculation of the maximum pressure exerted on the tube wall by the flow exhausting from the rocket nozzle. Thus, an understanding of the resultant flowfield in the launcher is needed. The demand for more powerful rockets coupled with the requirement for lightweight launchers has recently emphasized the need for understanding the flowfield. Test programs have been conducted to demonstrate the structural integrity of the launcher but have not used the opportunity to obtain flowfield information.

Contents

A highly underexpanded, supersonic, short-duration jet exhausting from a conical nozzle into a short tube with an inside diameter slightly larger than the nozzle exit characterizes the flowfield of the tube-launched rockets studied.

The experimental investigation included eight tests in which double-base, solid-propellant rockets were statically fired into the instrumented tubes and 48 tests in which dry nitrogen was expanded through a conical nozzle into instrumented cylindrical tubes of various lengths. Because the cold-gas tests formed the principal data base, the simulation of the actual rocket exhaust flows was an important consideration. A full-scale facility was designed to simulate the flowfield characteristics of the rocket exhaust. For both types of tests, the reservoir pressure varied rapidly with time, with the maximum value being as high as 350 atm. The test times were relatively short, e.g., approximately 10 msec for rocket firings and approximately 300 msec for the cold-gas tests. In addition, the tube-wall pressure distribution was obtained with a rocket accelerating through a tube launcher. To correlate the experimental data, a theoretical model based upon the method of characteristics was developed for the flowfield from the nozzle exit to a point just downstream of the impingement shock.

A sketch of the over-all flowfield based on the theory and substantiated by the acquired data is presented in Fig. 1. The axially symmetric, isentropic flow region is bounded by the inviscid jet boundary and the impingement shock. In actuality, this region may not be isentropic as assumed, if a strong boundary shock develops. However the assumption was found to be valid for the cases considered in the present investigation.

Once the jet boundary intersects the tube wall an impingement shock develops as a result of the turning of the gases along the wall. The flow aft of the impingement shock is highly rotational. From the launcher design viewpoint, this is the most important region because maximum wall pressure exists behind the impingement shock. The impingement shock projects to the tube axis as an apparent paraboloid surface. If the tube is sufficiently long, the impingement shock reflects from the axis back to the wall, producing a second pressure peak. Additional reflections take place depending upon the tube length.

Steady flow is assumed. However, in reality, the flow is not steady during rocket motor ignition or chamber pressure tailoff. The study concentrates on those data obtained during the "quasi-steady" flow which exists during the interval when the stagnation pressure is a maximum. The flowfield during this time interval is of interest from an engineering standpoint, because the stresses in the tube wall are at a maximum.

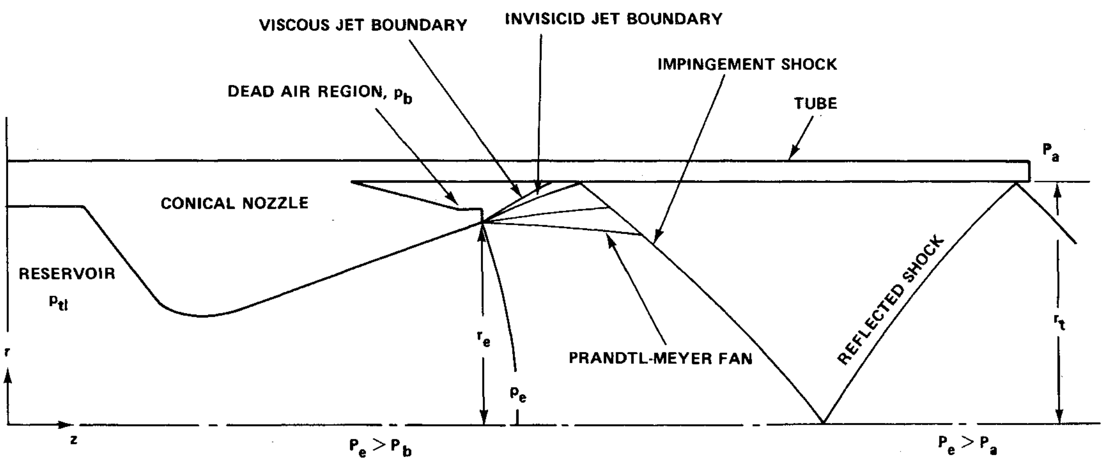

Fig. 1 Flow model of an underexpanded, axisymmetric, supersonic jet exhausting into a cylindrical tube.

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Index categories: Nozzle and Channel Flow; LV/M Structural Design; LV/M Subsystem Design.

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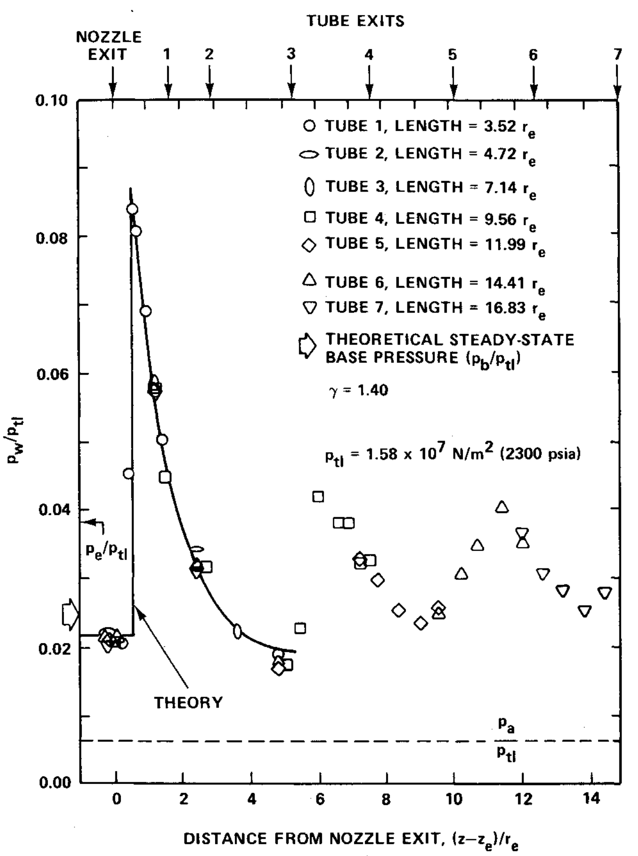

Fig. 2 Pressure distribution along the tube wall for the cold-gas flows.

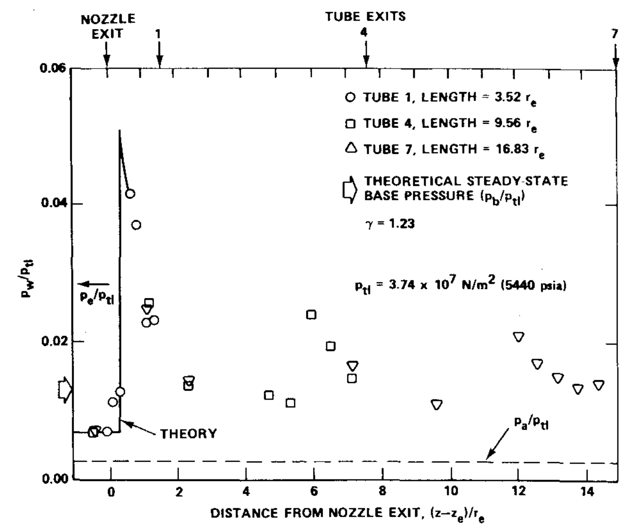

Fig. 3 Pressure distribution along the tube wall for the hot-gas flows.

The base pressure is a required input for the numerical routine which is used to calculate the isentropic, irrotational flow as it accelerates from the conical-flow region to the constant Mach number condition along the inviscid jet boundary. According to Korst et al.[1] the nozzle exhaust flow in the viscous shear layer that bounds the dead-air region and the physical geometric constraints dictate the equilibrium base pressure by mass transfer into or out of the dead air region. However, the response of the base pressure to changes in the boundary conditions requires an adjustment time[2] which depends on the governing mechanism, i.e., pressure perturbation or mass transfer. The transients in the present flowfields cause the dimensionless base pressure to be time dependent. As time increases, the dimensionless base pressure approaches the steady-state theoretical value. The theoretical equilibrium value was obtained using a numerical routine similar to that described by Addy,[3] but modified to calculate the base pressure for the sudden expansion of an axisymmetric jet into a shroud.

With the base-pressure : total-pressure ratio known, the Mach number and the initial flow turning angle can be determined. By using the methods of Ref. 4, one now can construct the characteristics network which defines the flowfield in the axially symmetric, isentropic-flow region. As the jet boundary encounters the wall it is turned parallel to the wall, creating an oblique shock wave. The oblique shock relations are used to calculate the Mach number behind the shock, the shock-wave angle, and the pressure rise across the shock. A new left-running characteristic is projected from the expansion wave into the potential flow region until the impingement shock is intercepted. The conditions upstream of the shock wave are determined by interpolation. The local shock-wave angle and the flow conditions downstream of the shock are then calculated.

The longest tube, which was 16.83-nozzle exit radii in length, was instrumented with 11 static pressure orifices. The pressure data from a single run was too sparse to adequately define the pressure profile along the tube wall. However, for the high stagnation pressures studied the viscous effects were insignificant and as one would expect for fully supersonic flow, the flowfield was independent of tube length. Experimental pressure distributions were obtained in each of the tubes and with various axial tube positions to provide pressures at additional positions relative to the nozzle-exhaust plane with the same instrumentation. By including all the pressures measured at a given stagnation pressure, sufficient data are available to define the pressure distribution.

The experimental static-pressure distribution along the wall is presented as a function of the nondimensionalized distance from the nozzle exit plane for the cold-gas flows and for the hot-gas flows in Figs. 2 and 3, respectively. Presented for reference are the pressure in the exit plane of the nozzle p_e, the theoretical value of the equilibrium base pressure p_b, and atmospheric pressure p_a (all nondimensionalized by division by the stagnation pressure p_{t1}). The locations of the nozzle exit plane, which is, of course, at $(z-z_e)/r_e = 0$, and of the exit planes of the various tubes are indicated along the upper axis. The static wall-pressure is significantly above atmospheric pressure along the entire tube length for the fully supersonic flows of Figs. 2 and 3.

Theoretical solutions of the flowfield from the nozzle exit plane to just downstream of the impingement shock, which were generated using the measured base pressure, are presented in Figs. 2 and 3. With the base pressure known, the theoretical solution provides an adequate prediction of the impingement-shock location and the static wall-pressures in this region. Furthermore, the theoretical pressure-distribution follows closely the experimental distribution (independent of base pressure), even though the theoretical steady-state (equilibrium) value and the experimental base-pressure differ. Thus, it is apparent that the base pressure depends on the total-pressure history and that the resultant flow in the tube is dependent on the ratio of the base pressure to the total pressure.

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