Rocket Exhaust Plume Dimensions

PHILIP ONNI JARVINEN* AND HARRY B. DYNER*
Mithras, A Division of Sanders Associates Inc., Cambridge, Mass.

Nomenclature

A/A^* = nozzle exit to throat area ratio
$C_F, C_{F\max}$ = vacuum thrust and maximum thrust coefficient, respectively
D = plume drag
$(D/q_\infty)^{1/2}$ = plume drag parameter
K_x = plume bow shock wave parabola constant
p_c = engine chamber pressure
q_∞ = freestream dynamic pressure
T = engine thrust
x = longitudinal distance downstream of nozzle exit plane
y = lateral distance measured from centerline of exhaust plume
γ = exhaust gas specific heat ratio

Introduction

THE size and shape of rocket exhaust plumes have been of interest for some time. The plume size is a result of the interaction between the atmosphere, the forward motion of the vehicle, and the jet flow from the rocket motor. Analytical techniques have been developed which describe the inviscid gasdynamic structure of high-altitude rocket plumes ($h > 100$ km). Hill and Habert[1] were the first to develop a simple theory to describe the size and shape of the initial portions of the plumes of expanding gases behind a missile in powered flight at altitudes above 100 km using the blast wave analogy. Their first-order theory predicts that the shock wave generated by a rocket exhaust plume is initially of parabolic shape with a parabola constant (or nose radius) whose magnitude varies directly as the square root of the ratio of plume drag to dynamic pressure, $(D/q_\infty)^{1/2}$. Therefore, $(D/q_\infty)^{1/2}$ may be used as a universal plume scaling parameter whose absolute magnitude determines the size of any rocket exhaust plume. Using conservation of axial momentum, they showed that the plume drag is related to the engine thrust, i.e., $D = T(C_{F\max}/C_{F\text{ex}} - 1)$, so the plume size of a missile with specific engine characteristics (γ, A/A^*) may be considered to vary as $(T/q_\infty)^{1/2}$. Alden[2] also noted that the dimensions of the contact surface for a jet issuing into a hypersonic stream varies as $(p_c/q_\infty)^{1/2} \propto (T/q_\infty)^{1/2}$. Moran[3] subsequently demonstrated by dimensional analysis that the entire structure of a jet issuing into a hypersonic ambient should scale as $(T/q_\infty)^{1/2}$. A comparison of theoretical predictions with optical observations of Atlas and Titan plumes was made[1] and showed excellent agreement in the nose radii (or parabola constant) and maximum radii of the plumes for values of $(D/q_\infty)^{1/2} \geq 5 \times 10^3$.

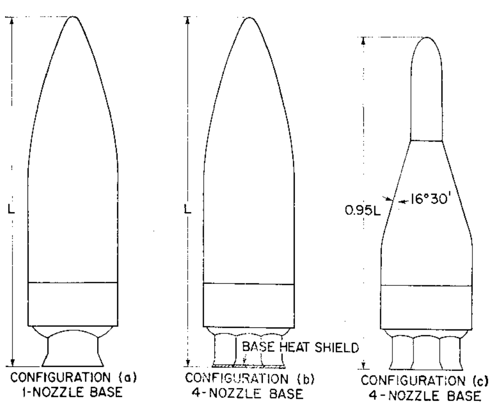

Fig. 1 Wind-tunnel model configurations.

Received August 11, 1969.
* Project Engineer, AeroSciences Department. Member AIAA.

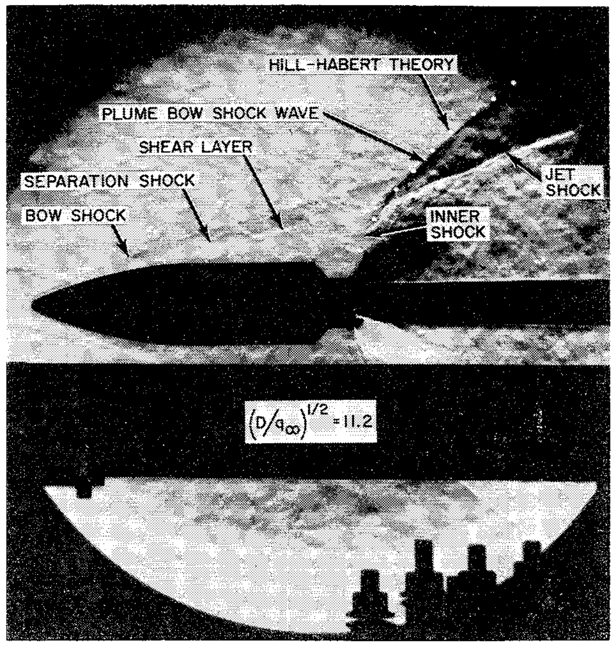

Fig. 2 Typical schlieren photograph of rocket exhaust plume—configuration a.

In this Note, rocket exhaust plume sizes determined from schlieren photographs taken during a series of wind-tunnel experiments at Arnold Engineering Development Center over the range $1.0 < (D/q_\infty)^{1/2} < 20$ are presented and compared with theoretical predictions of the Hill-Habert theory.

Wind-Tunnel Tests

Wind-tunnel tests were conducted in the Arnold Engineering Development Center von Kármán Facility (VKF) tunnel B at $M_\infty = 8.0$, in order to determine the plume induced flow separation pattern on several typical missile configurations. Schlieren photographs of rocket exhaust plumes issuing from the models shown in Fig. 1 were analyzed. The models used had one- and four-nozzle base configurations in combination with different forebody shapes. The model rocket exhaust was simulated with cold air delivered through hollow, single or multiple stings that passed through the models' conical nozzles. These stings were of conical shape inside the nozzles and retained their conical flow contour to a point far enough downstream to avoid any sting interference with the region of flow separation or nozzle flow interaction. Additional details of the model, sting support system, and jet flow and flow separation simulation are contained in Refs. 4 and 5.

Experimental Results

A typical schlieren photograph of the rocket exhaust plume is shown in Fig. 2. Features of the flowfield such as the separated flow region about the vehicle, the rocket exhaust plume, and the position of the plume bow shock wave are clearly visible. The photographs were scaled and it was found that the plume bow shock wave is parabolic. The photographic data for the location of the plume bow shock wave were reduced to yield the variation of parabola constant, $K_x = y^2/x$, with $(D/q_\infty)^{1/2}$ (Fig. 3). Parabola constants are plotted for single nozzle and multiple nozzle vehicles as well as for vehicles with different forebody shapes. Some results are shown for four engine configurations with a

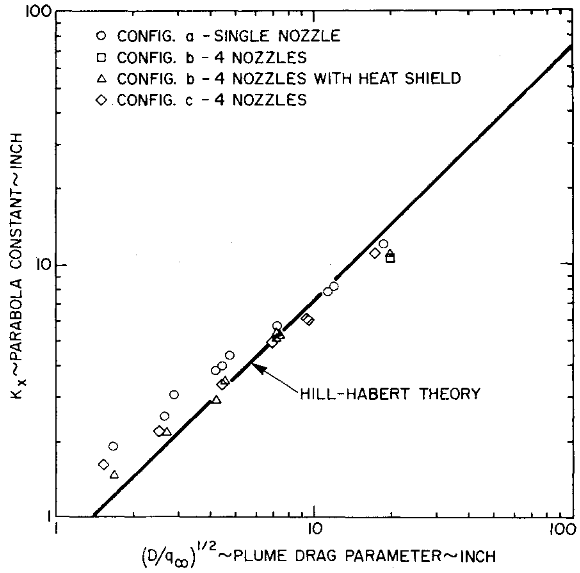

Fig. 3 Plume bow shock wave parabola constant.

heat shield which is located at the nozzle exit plane and fills the area between the four nozzles. The variation of the rocket exhaust plume parabola constant with plume drag parameter, $(D/q_\infty)^{1/2}$, is similar for all vehicle configurations that were analyzed. A usual approximation in theoretical calculations of the exhaust flow from multinozzle configurations is to replace the multinozzle configuration with a single "equivalent" nozzle. This assumption is supported by the limited data at $(D/q_\infty)^{1/2} = 20$, which shows nearly equal parabola constants for plumes from the four nozzle configuration (b) without heat shield and its single equivalent nozzle configuration (a).

The plume parabola constant predicted by the theory of Hill and Habert is shown in Fig. 3 for comparison. They predicted that

$$K_x = 0.57(\gamma_\infty/J_0)^{1/2}(D/q_\infty)^{1/2}$$

where γ_∞ is the ratio of specific heats in the ambient and J_0 is a function of γ_∞ tabulated in Ref. 6. For the experiments, J_0 is equal to 0.85. The plume shock position predicted by the Hill-Habert theory is noted in Fig. 2.

The excellent agreement between the Hill-Habert theory and the present experimental results for $(D/q_\infty)^{1/2}$ as low as 5.0 may be fortuitous, however, since their theory was originally developed for high-altitude plumes with $(D/q_\infty)^{1/2} > 5 \times 10^3$. With due consideration for this possibility, the excellent agreement with experiment indicates that the theory may be used to provide rapid and accurate estimates of dimensions of rocket exhaust plumes at significantly lower altitudes.

References

[1] Hill, J. A. F. and Habert, R. H., "Gasdynamics of High-Altitude Missile Trails," MC61-18-R1, Jan. 1963, Mithras Inc., Cambridge, Mass.

[2] Alden, H. L. and Habert, R. H., "Gasdynamics of High-Altitude Rocket Plumes," MC63-80-R1, July 1964, Mithras Inc., Cambridge, Mass.

[3] Moran, J. P., "Similarity in High-Altitude Jets," *AIAA Journal*, Vol. 5, No. 7, July 1967, pp. 1343–1345.

[4] Adams, R. H., "Wind-Tunnel Simulation of Rocket Vehicles in Flight with Two-Phase-Flow Rocket Plumes," *Journal of Spacecraft and Rockets*, Vol. 4, No. 4, April 1967, pp. 518–524.

[5] Alpinieri, L. J. and Adams, R. H., "Flow Separation Due to Jet Pluming," *AIAA Journal*, Vol. 4, No. 10, Oct. 1966, pp. 1185–1186.

[6] Sakuari, A., "On the Propagation and Structure of Blast Waves, I.," *Journal of the Physical Society of Japan*, Vol. 8, No. 5, 1953.

Gas Effects in Attitude Control Systems

W. Juran,* E. B. Zwick,† and R. L. Moffett‡
The Marquardt Corporation, Van Nuys, Calif.

DIFFUSION through a tank bladder is a slow way of saturating a body of propellant with gas if the propellant is quiescent. However, any motion due to sloshing or thermal convection increases the solution rate. When bladders are made of materials (such as Teflon) with different permeabilities for pressurant gas and propellant vapor, free gas bubbles can grow in the propellant by osmosis. When the propellant is saturated with pressurant gas, the partial pressure of the pressurant gas may be lower on the liquid side of the bladder than on the gas side. This partial pressure difference arises because the bladder tends to prevent the propellant vapor from making as big a contribution to the total pressure on the gas side of the system as it does on the liquid side of the system (where its vapor pressure is exerted). (The amount of gas in the propellants can be minimized by techniques such as using aluminum bladders. This was done on the Lunar Orbiter vehicles.[1] The Surveyor vehicles did not use this type of gas barrier but a program of ground tests was conducted to insure that gas effects would not interfere with obtaining mission objectives.[2])

During a mission, propellant pressures and temperatures can change because of blowdown operation, temperature changes in the environment, thermal soakback from the rocket engine, or because propellants flowing through a rocket engine meet lower pressures and higher temperatures between the engine inlet and the rocket injector orifices. When the propellant temperature and pressure conditions change so that the amount of gas in the propellant exceeds the equilibrium amount, some dissolved gas may come out of solution.

When propellants contain dissolved gas, free gas bubbles can form in the propellant feed lines and propellant valve bodies during priming. Exposure of the gas-containing propellants to the low pressure and turbulent flow during priming generates free gas, which may not redissolve when priming is complete. If the feed lines are not evacuated before priming, the trapped gas contributes to the free gas in the system after priming.

Free gas can affect attitude control system (ACS) operation in a number of ways: combustion pressure oscillations, transient changes in chamber pressure and thrust, long ignition delays, and shifts in pulsing and steady-state performance parameters.

Analytical and Experimental Results

Figure 1 shows two types of special test equipment used to evaluate gas effects on ACS operation. Special tanks with

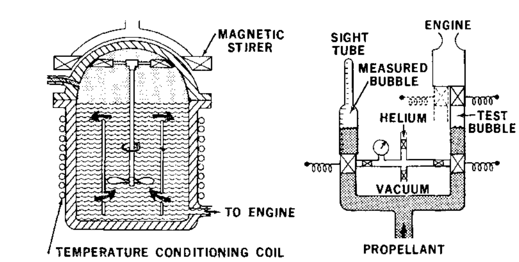

Fig. 1 Test apparatus.

Presented as Paper 69-434 at the AIAA Propulsion Joint Specialist Conference, U.S. Air Force Academy, Colo., June 9–13, 1969; submitted June 18, 1969; revision received August 22, 1969.

* Manager, Systems Engineering, Applied Physics Laboratory Computing Services, Division of The Marquardt Company.

† Consultant. Member AIAA.

‡ Project Engineer, Propulsion Division.