Rocket Plume Testing in Ground Test Facilities

R. J. SCHULZ* AND R. C. BAUER*
ARO, Inc., Arnold Air Force Station, Tenn.

Four basic types of ground test facilities are reviewed briefly with respect to the degree of simulation of flight environment that they can provide for propulsion engine exhaust plumes. The importance of simulating the freestream in relative motion to a rocket exhaust plume is estimated by determining analytically the effect of the freestream on the location of the "inviscid" exhaust plume boundary by a Korst-type analysis. Results show that a considerable portion of the rocket plume is affected primarily by the pressure in the region of plume induced separated flow. Thus, in many cases it is possible to simulate flight conditions without providing a coflowing supersonic stream by setting the test cell static pressure equal to the average pressure in a plume induced region of flow separation corresponding to a flight condition of interest. Parameters characterizing the exhaust nozzle and exhaust gases which affect flow separation on a missile afterbody are indicated, and the utility, in fact the necessity, of small scale plume testing is discussed.

Nomenclature

D_e = missile base diameter
H_∞ = flight altitude
H_{eff} = effective pressure altitude
l_s = length of separated flow region
L = missile length from base to shoulder
M_∞ = freestream Mach number
M_{je} = rocket nozzle exit Mach number
M_{j3} = rocket plume Mach number downstream of impingement
P_c = rocket chamber pressure
P_∞ = freestream static pressure
P_{je} = rocket nozzle exit static pressure
P_{j3} = rocket plume static pressure downstream of impingement
P_b = pressure in PIES region
u_∞ = freestream velocity
$\bar{X}, \bar{Y}$ = coordinate axes, body referenced
x, y = coordinate axes, shear layer referenced
γ_{je} = ratio of specific heats of exhaust gas
η = similarity coordinate
θ = flow angle
θ_{ext} = flow angle of attached supersonic stream
θ_{je} = rocket nozzle divergence half-angle
σ = shear layer growth constant

Subscripts
1 = undisturbed flow along missile afterbody
2 = deflected flow outside separation region
3 = flow downstream of plume—freestream impingement point

Introduction

THE primary purpose of this Paper is to discuss the problem of obtaining data on rocket exhaust plumes, in particular, the requirements for achieving an accurate ground test simulation of the inflight environment of exhaust plumes.

Figure 1 is a simplified schematic of a rocket propelled missile with an underexpanded exhaust plume. The flowfield surrounding the plume is a supersonic stream, which after separating from the missile body, impinges upon the exhaust plume. Compression waves are generated by the missile forebody which modify the freestream, hence, the exhaust plume. The character of the plume-induced external separation (PIES) region depends upon the missile forebody boundary layer, and the shear layers that exist along the boundaries of the PIES region. The mixing layer along the plume boundary can be either laminar or turbulent and its chemical state depends upon the reactions which may be occurring in this shear layer; that is, whether the reactions can be considered frozen, proceeding at an equilibrium rate, or at a "finite" rate. The nature of both shear layers can rapidly change as the missile traverses its trajectory since pressure and temperature of the freestream vary with flight altitudes and Mach number.[1]

Up to the present time, knowledge of large rocket plumes has come from radar studies, photographic studies, and plume radiation studies of full scale in-flight rockets. In these studies the plume often appears as a point source for radiation and details of the plume are difficult to assess. Another important factor is that the aerodynamic variables affecting the plume are not controllable. Hence, at present, the ground test facility offers the most practical way to gather a large portion of detailed data on rocket plume properties.

The basic ground test facilities in which the environment of rocket engines and exhaust plumes are simulated to various degrees of approximation are: a) static test cell, b) vacuum chamber, c) wind tunnel, and d) centerbody tunnel. These test cells can be operated in either steady-state or transient modes. Clearly, centerbody tunnels and wind tunnels are closely akin since, basically, the centerbody tunnel is a wind tunnel designed for operation with a centerbody installed. Nevertheless, a distinction will be made in this paper between the usual wind tunnel and the centerbody wind tunnel. It should be pointed out that in the discussions which follow and throughout the

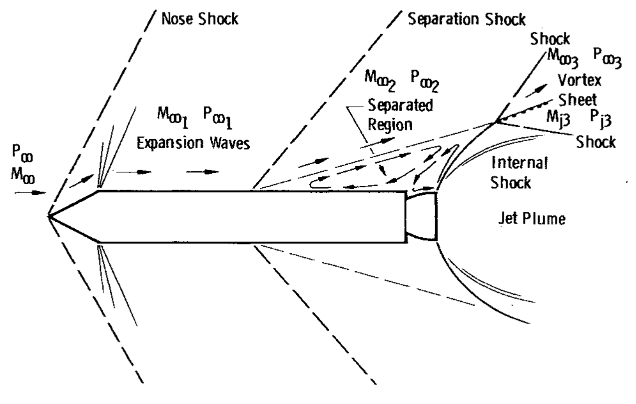

Fig. 1 A sketch of jet plume induced flow separation.

Presented as Paper 72-1072 at the AIAA/SAE 8th Joint Propulsion Specialist Conference, New Orleans, La., November 29–December 1, 1972; submitted April 4, 1973; revision received August 28, 1973. This research was sponsored by the Arnold Engineering Development Center.

Index category: Rocket Engine Testing.

* Research Engineer. Member AIAA.

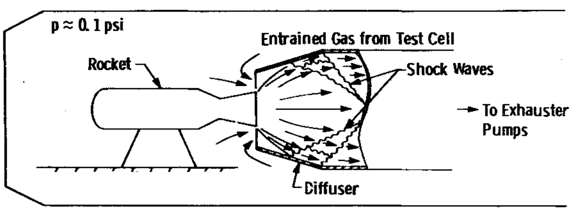

Fig. 2 Rocket installation in a static test cell.

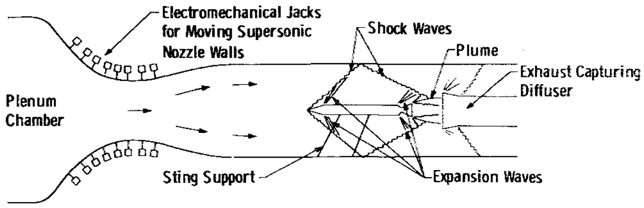

Fig. 4 Rocket installation in a supersonic wind tunnel.

remainder of this Paper, the comments and results apply specifically to plumes at zero angle of attack.

Figure 2 is a schematic of a static test cell. In this kind of test cell, examples of which have been documented and described in detail,[2] a rocket is mounted on a thrust stand and an exhaust capturing diffuser is positioned near the exit plane of the rocket nozzle. The performance of the engine, including its thrust, integrity, and operational life, is obtained by such testing. The only requirement for simulation of flight environment is that a representative static pressure, and often temperature, be maintained in the test cell prior to and during firing. The ejector action of the rocket plume impinging on the diffuser wall maintains the cell pressure. Plume properties are usually not of interest in this kind of testing unless they are nozzle exit plane plume properties. Existing test cells of this type can be used to test rockets of all sizes.

Figure 3 shows the schematic of a small rocket exhausting into a vacuum chamber. This kind of testing is used principally to study characteristics of plumes expanding into very high altitude conditions, typically, to study inviscid plume structure or radiation characteristics. Since the mass pumping rates are relatively small for these kinds of facilities, rockets which are tested must be small in order to test under invariant conditions. Examples of these types of test cells have also been documented.[3]

Figure 4 is a schematic of a strut-mounted rocket in the test section of a supersonic wind tunnel. In wind tunnels a range of rocket flight environments can be simulated fairly accurately. Typical wind-tunnel facilities are well documented in the literature.[4] For a model of a given size, tests conducted in conventional wind tunnels are usually more costly on a per hour basis than tests in the other type facilities considered. Expenses increase with tunnel size because of the equipment and operating costs required for pumping large volume flows. Also, disturbances which reflect off the wind-tunnel walls must be accounted for in testing, Fig. 4, and, in many cases, the exhaust plume must be captured by a diffuser. However, usage of the wind tunnel enables the effects of the supersonic stream on the plume and the characteristics of the PIES region to be measured in detail.

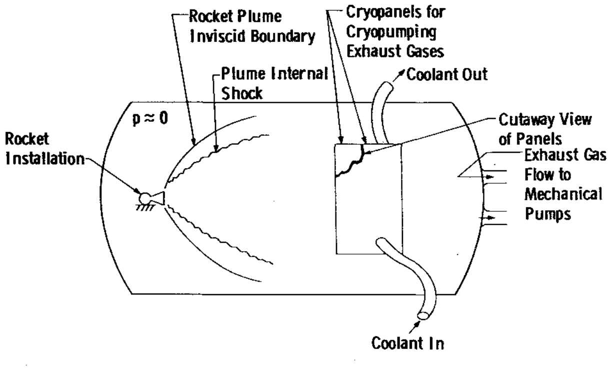

Fig. 3 Rocket installation for simulating vacuum conditions.

Figure 5 is a schematic of a centerbody tunnel. In this configuration, a propulsion engine is mounted in the base of a centerbody of a supersonic nozzle. The nozzle and centerbody combination are designed to produce uniform disturbance-free supersonic flow (or tailored flow) at the plane of the base of the centerbody.[5]

This test technique eliminates disturbances from the reflected bow shock or from support struts which occur in the conventional wind tunnel. The rocket plume interacts with the external freestream in the same way as in the supersonic wind tunnel, causing disturbance shock waves which reflect off the tunnel walls and return to disturb the plume. With this configuration, testing is restricted to one freestream Mach number for each nozzle-centerbody configuration, although it is possible to vary the freestream pressure. Because this testing configuration usually is an open circuit, the exhaust gases do not affect the air supply, thus, a diffuser is not needed to capture the plume.

For an equivalent model size, tests conducted in annular nozzle facilities are less expensive per test hour than in conventional wind tunnels. The annular nozzle tunnel often represents a good compromise between the cost of providing external flow and the range of data that can be obtained.

At this point, it is important to assess if it is possible to extend current or existing test facilities and test methods to improve the simulation of the flight environment of propulsion engine exhaust plumes. Specifically, is it possible to improve and extend the range of simulation of flight environment in the static test cell and in vacuum chambers, which do not, in general, have the capability for providing the supersonic flow which surrounds a missile in flight? In addition, if small scale model testing must be done, what is the utility of such testing, since in many practical cases of interest, finite rate chemical reactions and difficult to measure viscous-inviscid interactions preclude direct scaling of the test results to interpret full scale phenomena?

As a first step to answering the first questions, it is pertinent to determine, for typical flight conditions, the effects of the external supersonic stream on the gross inviscid structure of the rocket plume. Specifically, is it possible that there is a representative static pressure that can be rationally applied to enable external flow simulation testing in a static test cell without any provision of external flow? This question was approached in a study conducted at the Engine Test Facility of Arnold Engineering Development Center in 1968–1969.

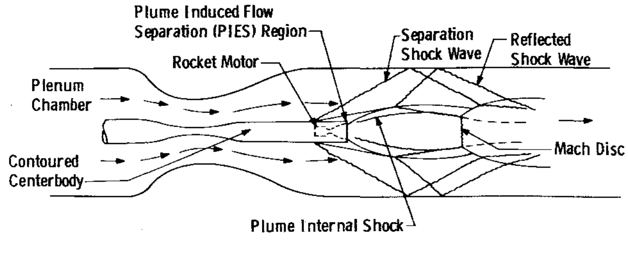

Fig. 5 Rocket installation in a centerbody tunnel.

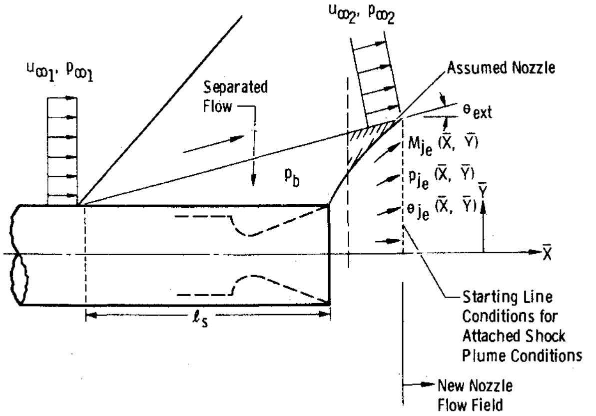

Fig. 6 Starting line for calculating the exhaust plume downstream of external flow impingement.

Analysis of External Flow Effects on Inviscid Plume Structure

A. The Method of Analysis

The basic analytical approach is to define the separated region by use of the Korst base pressure theory, and then apply the method of characteristics solution technique to determine the downstream plume structure. Sufficient details of this analysis are presented elsewhere[6] so that only the key features are given here. The analysis can account for missile afterbody and base geometry, missile flight trajectory, exhaust gas chemical composition, and propulsion nozzle geometry. Clustered nozzles and missile flight angle of attack can be treated by a strip technique. The analysis was applied to a simple rocket configuration consisting of a cone-cylinder having a total length to diameter ratio of 10. The nose cone length was one body diameter so that it was reasonable to assume the flowfield over the aft portion of the missile had recovered back to the undisturbed freestream conditions. The propulsion nozzle was assumed to be a Mach Number 3.0, 15° half-angle conical nozzle with an exit diameter equal to the missile diameter. Rocket combustion chamber conditions of 500 psia and 3000°R were assumed and the exhaust gas composition was specified by two different values of specific heat ratio, γ, and molecular weight, W. Two flight trajectories representative of sounding rockets, and trajectories of constant Mach number were considered. In all cases, the trajectories were defined by flight Mach number as a function of pressure altitude

The pressure in the PIES region, P_b, was assumed to be constant and equal to a value yielded by an empirical specification developed[6] by correlating very high Reynolds number plume induced boundary-layer separation data. With the pressure, P_b, given by the empirical specification at each selected point along a flight trajectory, the Korst component-type base pressure theory was applied to compute the extent or length of the separated flow region. For a selected flow separation point the "restricted" base pressure theory of Korst was applied; this means that the missile forebody boundary layer is neglected and the mixing zone velocity profiles are similar and defined by the following equation

$$u(\eta)/u_{\infty}(x) = \tfrac{1}{2}(1+\operatorname{erf}\eta) \tag{1}$$

where

$$\eta = \sigma y/x \tag{2}$$

and erf (η) represents the classical error function. The Chapman-Korst isentropic stagnation criteria was used to determine the stagnating streamline in the recompression process. This criteria assumes that the total pressure (upstream of the recompression) on the streamline that stagnates must equal the maximum static pressure in the recompression zone. The total pressure distribution in the mixing zone upstream of the recompression zone can be computed by use of Eq. (1) and the assumption that the static pressure distribution through the mixing zone is constant and equal to the pressure, P_b, in the PIES region. In the original theory by Korst the maximum static pressure in the recompression zone was determined entirely by the turning of the inviscid flowfield thus assuming the mixing process has no effect on the peak static pressure determined by the recompression process. This is a good assumption only if the inviscid flowfield is uniform as would be true for two-dimensional flow over a backward facing step. For an axisymmetric plume the inviscid flowfield is highly nonuniform and experimental results[7] show the maximum static pressure in the recompression zone is significantly influenced by the mixing process. In addition, for the type of inviscid nonuniform flow existing in an axisymmetric plume, the maximum static pressure in the recompression zone is determined by the turning of the inviscid flow conditions along the plume-edge of the mixing zone. Comparison with experimental data shows the plume-edge of the mixing zone can be assumed to be where $u(\eta)/u_{\infty}(x) = 0.9846$ which corresponds to $\eta = 1.53$. Thus, the peak recompression pressure established by the jet plume was estimated by first computing the jet plume flowfield by the method of characteristics then computing the size of the mixing zone based on an Abramovich mixing rate, then superimposing the mixing zone on the inviscid flowfield using conservation of momentum to establish the proper orientation, and then using the inviscid flow conditions along the plume-edge of the mixing zone and two-dimensional shock wave theory to determine the peak recompression pressure. The external stream is assumed to be uniform adjacent to the mixing zone so the peak recompression pressure for this stream is not influenced by the mixing process. The final condition imposed on the recompression process is that the peak pressures established by the external stream and the jet plume must equal. This condition is satisfied by the proper slip line angle.

Having determined the peak recompression static pressure, it was then possible to determine the stagnating streamlines by applying the previously discussed isentropic criteria. The dividing streamline location in the mixing zone was determined by applying the conservation of mass flow and momentum to the mixing process as was done by Korst. The net mass flow being pumped into or out of the separated region by each

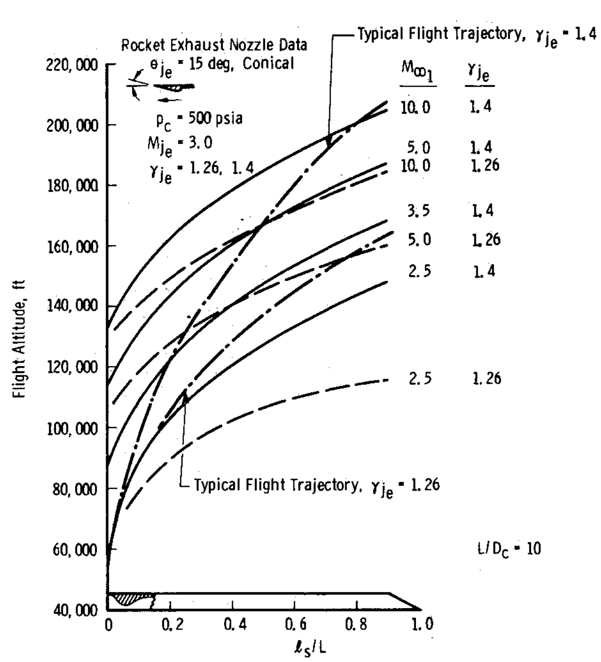

Fig. 7 Nondimensionalized Flow separation distance as a function of flight trajectory, γ_{je}, and $M_{\infty 1}$.

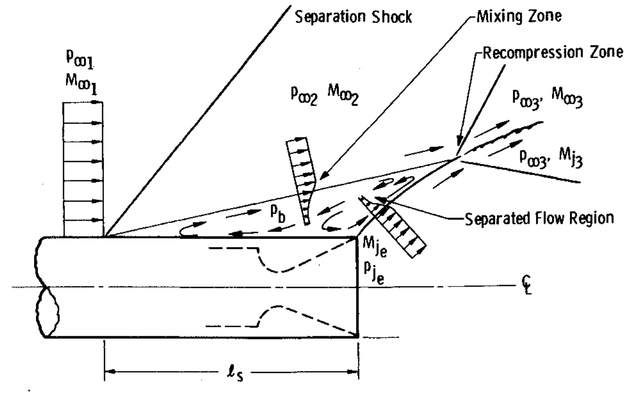

Fig. 8 A simplified sketch of plume induced external flow separation.

stream is that contained between the dividing streamline and the stagnating streamline. Thus a mass balance is computed for each assumed separation condition. The correct external flow separation point is determined when the total net mass flow into the PIES region is zero.

After obtaining the extent of external flow separation, the point of impingement of the external stream on the plume is known. This includes the angle of impingement and the external stream pressure and Mach number. Then at this location, an equivalent supersonic nozzle flowfield can be defined with an attached supersonic uniform stream defined externally to the nozzle. Figure 6 is a schematic which shows the newly defined flowfield. A rotational method of characteristics solution can be made of the resulting downstream field using these initial conditions and the attached external supersonic stream boundary conditions. A computer program developed by the Lockheed Company[8] was utilized in this study to perform these attached flow solutions. This program uses Newtonian impact theory to describe the pressure distribution on the plume boundary. Thus, the remainder of the supersonic plume inviscid structure can be computed downstream of the point of impingement with the external stream. Then, the over-all plume structure, from rocket nozzle exit plane to far downstream, can be constructed for a given flight condition. From this, the changes in plume inviscid structure can be estimated by comparing the plume computed with this method to the plume computed expanding into a quiescent region of the same static pressure, P_∞, but with Mach number, M_∞, of zero. It should be noted that a similar analysis of jet plume flow separation using Korst base pressure theory and an empirical separation pressure ratio function has been reported.[9,10] These investigations were not directly concerned with the prediction of plume structure changes due to external flow.

B. Results of the Analysis

Figure 7 shows the theoretical nondimensionalized separation distance, $l_s/_L$, as a function of flight Mach number and altitude for the chosen missile geometry for two different exhaust gas compositions, as indicated by the values of γ_{je}. Included in Fig. 7 is the separation distance, l_s, as a function of γ_{je} for both constant Mach number trajectories and for a typical sounding rocket trajectory. Two important effects that are immediately obvious from this figure; first, the exhaust gas with the lower γ_{je} induces a greater extent of flow separation at a given flight condition, and second, it is possible to define a trajectory for which no flow separation will occur. This trajectory is defined in Fig. 8, by the points of intersection of the constant Mach number lines with the ordinate, for a given γ_{je}.

The effects of the impinging flow on the plume inviscid structure were computed by the method previously described. In Fig. 9 the trend of these effects can be determined by comparing the jet inviscid boundaries and internal shock

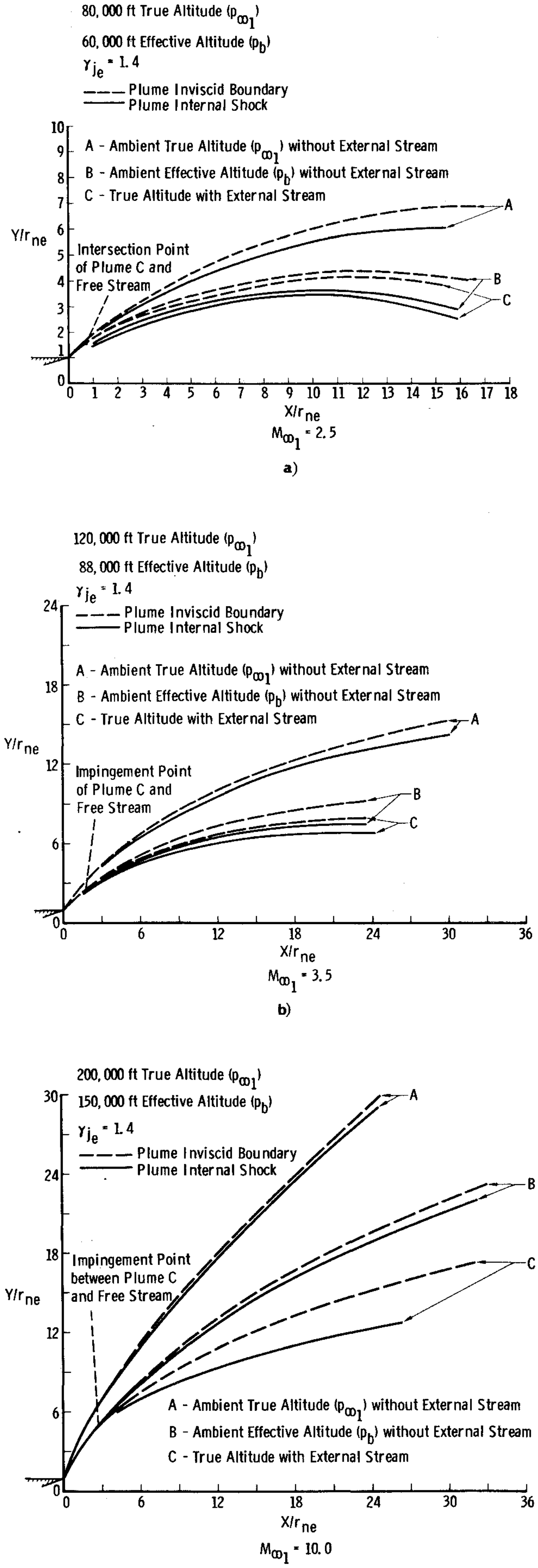

Fig. 9 Effect of external stream on rocket plume structure.

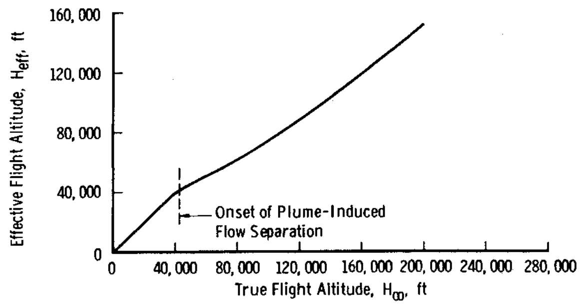

Fig. 10 Effective flight altitude vs true flight altitude for a sounding rocket trajectory.

locations for three different flight conditions with and without consideration of external flow. The plume structure indicated by A in Fig. 9 is the plume which expands into ambient conditions, $P_\infty = P_{\infty 1}$, and $M_{\infty 1} = 0$. There is no impinging stream. The plume indicated by B in Fig. 9 is the plume which would expand into an ambient condition defined by $P_\infty = P_b$ where P_b is given as a function of $P_{\infty 1}$ and $M_{\infty 1}$. That is, it is given by the empirical separation pressure ratio function. Plume B is the plume which would occur if only the pressure in the external separated region determined plume structure. In Ref. 6, P_b was used to define an effective ambient altitude for each flight condition experienced by the rocket as it traversed its trajectory. The plume indicated by C in Fig. 9 is the plume geometry which results from both the effect of an effective ambient altitude, P_b, and the impingement of a supersonic external stream with its inviscid boundary. By comparing these plumes, it was concluded that for low (less than 80,000 ft) altitude and low (less than $M_{\infty 1} = 2.5$) Mach number flight the rocket exhaust plume structure is primarily determined by the pressure in the PIES region. The impingement of the supersonic stream causes relatively minor changes in plume structure, that is, B and C compare closely in this range of flight conditions. For higher flight altitudes and Mach numbers, the effect of impingement on the plume structure increases as shown in Fig. 9. Thus, the effect of the pressure in the PIES region and the effect of the external stream impingement are both important (in these cases) in establishing plume structure. The effective ambient altitude is illustrated in Fig. 10 for the missile traversing the sounding rocket trajectory.

Summary and Conclusions

At present, the most important need is to obtain experimental data for this phenomena in order to verify the results and improve the theory. Some work of this kind has been done by the Army Missile Command for the low Mach number, transonic flow region.[11–14] The conclusions of the present study which pertain to ground test flight simulation techniques are as follows.

As a first approximation for simulating the environment for plume tests where the inviscid plume properties in the flowfield near the nozzle exit are desired, it is sufficient to test in a static environment with the pressure in the test cell corresponding to the effective altitude for the given flight conditions.

For tests in which the entire plume flowfield or a substantial part of the flowfield down to an including the first Mach disk is required for study, and for flight conditions of a rocket where external flow separation cannot be simulated only by static pressure, the proper simulation of the environment is achieved by providing the external coflowing supersonic stream about the engine.

Care must be taken in model studies using cold jets to simulate hot rocket exhaust plumes or in using hot jets whose chemical composition is substantially different from the full scale plume. The study showed that the ratio of specific heats, γ_{je}, of the exhaust jet and the nozzle exit angle, θ_{je}, have large effects on the separation phenomena.

No general criteria can be developed for predicting the extent of flow separation and each particular case requires an analysis. Such an analysis can be made using the Korst base pressure theory as developed in Refs. 6, 9, and 10. In addition, pertinent modifications to the simple Korst theory would be required in the method of Ref. 6, for example, to account for the effects of thick boundary layers ahead of the separation point and nonconstant pressure in the PIES region, in order to arrive at valid conclusions for particular cases which are different from the ones treated in Ref. 6.

Finally, in practice, there is strong justification for conducting small scale investigations since this is the only source of detailed information that can be used to verify theoretical ideas and analytical methods. Once these analytical techniques are well verified for small scale phenomena they can be applied to predict full scale effects. In this sense, a rather elaborate but physically perceptive analysis may be considered a formalized scaling procedure.

References

[1] Yoshihara, H., "Gas Dynamics of Rocket Exhaust Plumes," Rept. GDA 63-1381, ERR-AN-440, 1963, General Dynamics, New York.

[2] Goethert, B. H., "High Altitude and Space Simulation Testing," ARS Preprint 2060-61, 1961, ARS Space Flight Report to the Nation, New York Coliseum, New York.

[3] Purello, C. H., Harden, R. D., Heckert, M. V., and Brown, K. R., "An Inventory of Aeronautical Ground Research Facilities—Vol. III—Structural and Environmental Facilities," CR-1876, Nov. 1971, NASA.

[4] Purello, C. J., Hardin, R. D., Heckert, M. V., and Brown, K. R., "An Inventory of Aeronautical Ground Research Facilities—Vol. I—Wind Tunnels," CR-1874, Nov. 1874, Nov. 1971, NASA.

[5] Page, R. H., Dixon, R. J., and Przirembel, C. E. G., "Apparatus for Studies of Supersonic Axisymmetric Base Flow," *Proceedings of XVIIth International Astronautical Congress*, International Astronautical Federation, Madrid, Spain, 1966, pp. 347–353.

[6] Schulz, R. J., "Rocket Exhaust Plumes in a Separated Supersonic External Stream," M.S. thesis, Dec. 1970, Department of Mechanical Engineering, University of Tennessee Space Institute, Tullahoma, Tenn.

[7] Bauer, R. C., "Theoretical Base Pressure Analysis of Axisymmetric Ejectors without Induced Flow," AEDC-TDR-64-3, Jan. 1964, Arnold Engineering Development Center, Tullahoma, Tenn.

[8] "Users Manual—Description of a Digital Computer Program for Nozzle and Plume Analysis by the Method of Characteristics," TM-54/20/-108, Dec. 1966, Lockheed Missiles and Space Co., Sunnyvale, Calif.

[9] Fong, M. C., "An Analysis of Plume-Induced Boundary-Layer Separation," *Journal of Spacecraft and Rockets*. Vol. 8, No. 11, Nov. 1971, pp. 1107–1114.

[10] Fong, M. C. and Ehrlich, C. F., Jr., "Propulsion Effects on Aerodynamic Characteristics of Lifting Reentry Vehicles," AFFDL-TR-70-12, March 1970, Air Force Flight Dynamics Lab., Wright-Patterson Air Force Base, Ohio.

[11] Addy, A. L., "Analysis of the Axisymmetric Base-Pressure and Base-Temperature Problem with Supersonic Interacting Freestream—Nozzle Flows Based on the Flow Model of Korst, et al. Part II: A Comparison and Correlation with Experiment for Cylindrical Afterbodies," Rept. RD-TR-69-13, Dec. 1969, Army Missile Command, Redstone Arsenal, Ala.

[12] Rubin, D. V., "A Transonic Investigation of Jet Plume Effects on Base and Afterbody Pressures on Boattail and Flare Bodies of Revolution," Rept. RD-TR-70-10, Oct. 1970, Army Missile Command, Redstone Arsenal, Ala.

[13] Burt, J. R., Jr., "An Experimental Investigation of the Effects of Several Rocket Plume Simulators on the Pressure Distribution of a Body of Revolution at Freestream Mach Number of 0.9 to 1.2," Rept. RD-TR-70-23, Nov. 1970, Army Missile Command, Redstone Arsenal, Ala.

[14] Wu, J. M. and Aoyama, K., "Transonic Flow Fields Around Various Bodies of Revolution Including Preliminary Studies on Viscous Effects with and without Plume," Rept. RD-TR-71-12, May 1971, Army Missile Command, Redstone Arsenal, Ala.

[15] Peters, C. E. and Phares, W. J., "The Structure of Plumes from Moderately Underexpanded Nozzles," AIAA Paper 70-229, New York, 1970.