Separation and Stability Studies of a Convergent-Divergent Nozzle

E. H. MILLER* AND D. MIGDAL†
Grumman Aerospace Corporation, Bethpage, N. Y.

Operation of a convergent-divergent (C-D) nozzle at low-pressure ratios can cause the internal flow to separate and possibly be unstable. A test program was performed using a 1.2 area ratio scale model nozzle to determine the nature of the flowfield. Secondary flows of 0 and 2% were introduced into the nozzle. Results of these tests are as follows: 1) High response pressure transducer readings internally along the nozzle lip provided no indication of a moving shock, either axially or circumferentially. Pressure fluctuations were greatest at a location near the point of separation. The maximum amplitude was small, 3.5% of the total pressure. 2) Hysteresis tests were run with the possibility of their uncovering a clue as to unstable separation in a more elastic model. No hysteresis was observed. 3) Secondary flows had little effect on the maximum amplitude of the pressure fluctuation. Its main influence was to move the shock system slightly upstream in the nozzle.

Introduction

SUPERSONIC aircraft require convergent-divergent nozzles for optimum performance at high Mach numbers and corresponding large nozzle pressure ratios (total/ambient). Sometimes it is necessary to operate such a nozzle at low-pressure ratios with the accompanying possibility of flow separation. The exact nature of the separation, i.e., whether it is stable or not, could affect an aircraft's control characteristics. Early work[1] demonstrated asymmetric separation in large area ratio nozzles. Irregular fluctuations of the exhaust jet were found[2] with nozzles having design pressure ratios greater than 4.0 when operating at low nozzle pressure ratios (NPR). Later, reference was made[3] without supporting data to unsteady and asymmetric shock systems in a region where the shock sits just downstream of the throat. Additional data have been presented on aeroelestic instability of ejector nozzles at low nozzle pressure ratios.[4] This data indicated that instability in an elastic nozzle might be deduced with a rigid model if the latter exhibited hysteriesis, i.e., a dual value of internal nozzle wall pressures depending on the direction in which a given NPR is approached. Instabilities were also encountered during full scale engine tests with a low base drag ejector nozzle.[5] Much separation data have been generated and correlated for large area ratio nozzles used in rockets.[3] However, little data are available[2] for continuous convergent-divergent nozzles at low area ratios (exit/throat). The test program discussed herein was aimed at providing data in this range. A 1.2 area ratio convergent-divergent nozzle with an 11°45′ exhaust half angle was investigated. Modern aircraft nozzles with a high-temperature afterburner exhaust may utilize cooling flow introduced upstream of the throat. To determine the effect of this cooling flow on nozzle separation performance, 0 and 2% secondary flows were used.

Discussion

I. Test Procedure

The nozzle was tested without wind-tunnel flow (static conditions) at FluiDyne Corporation, Minneapolis, Minn. The nozzle pressure ratio was determined by use of a Heise gage reading for the nozzle total pressure. Manometer boards were used to measure the nozzle pressures for the hysteresis tests where accuracy was important. As soon as the manometer board readings stabilized, Polaroid photographs were taken of the boards to record the nozzle pressures. During hysteresis testing, the nozzle pressure ratio was increased and then decreased. For the transient tests, a transducer-oscillograph system was substituted for the manometer boards to record pressures. The transducers were calibrated initially with the manometer boards. Room temperature air, used for the nozzle flow was measured by means of an ASME nozzle located upstream of a choke plate and a series of screens. A set of total pressure probes were located upstream of the nozzle entrance. These were used to measure flow as a function of nozzle total pressure when the secondary passage was closed off. The secondary flow was measured by a series of orifices and static pressure probes upstream of the choke plate. The difference between the secondary and total flow was assigned to the primary nozzle.

Selected shadowgraph pictures were taken of the nozzle exhaust to determine the stability of the jet and the progression of the shock out of the nozzle, i.e., downstream, with increasing nozzle pressure ratio.

II. Model and Instrumentation Description

The convergent-divergent test nozzle was built of steel and had an area ratio of 1.2. The internal shape was a circular arc throat joined to a conical exit section with a half angle of 11°45′. At a stagnation pressure of two atmospheres, the flow Reynolds number based on throat diameter was $2.12 \times 10.^6$ A liner was located upstream of the nozzle throat to provide secondary flow. The assembly is shown in Fig. 1. Twelve pressure taps were located in the nozzle to record the pressure data (see Fig. 2). Four were located axially at 0°; another four were 180° apart at the same axial location. The last four were staggered between the others, thus providing eight axial stations. The pressure taps were connected to Hidyne transducers which were selected for their small volumes (0.001 in.^{3}) and, thereby, their good response. The transducers were referenced to ambient pressure and were connected to an amplifier. No filters were used in the system (see schematic of test setup, Fig. 3). The line lengths to the transducers were on the order of 10 in. (see Fig. 4). Internal diameters of the tubing ranged from 0.052 in. at the stripper tube to 0.03 in. at the pressure tap. They were hooked up to a Consolidated Electrodynamics Corporation

Received March 7, 1969; revision received August 29, 1969. The authors wish to acknowledge the work performed at FluiDyne Engineering Corporation by O. Lamb, J. Grunnet, and D. Kamis for their assistance in conducting and evaluating the experimental test program.

* Propulsion Advanced Development Project Engineer. Member AIAA.

† Propulsion Advanced Development Project Engineer. Member AIAA.

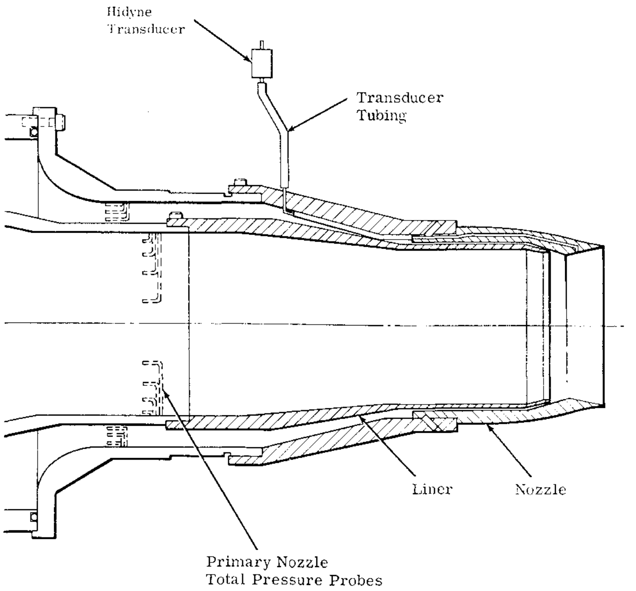


Fig. 1 Nozzle test schematic.

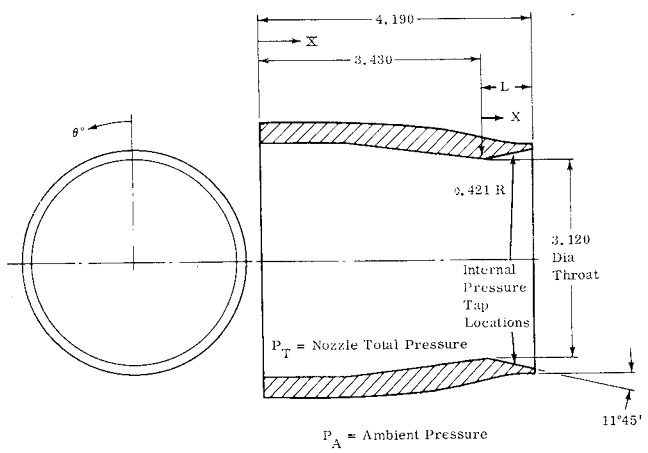
(CEC) oscillograph. Data were recorded on light-sensitive paper which was moving at a speed of 4 in./sec. The frequency response of the electrical system with the damping was 0–350 Hz. It was estimated that the pressure readings were accurate to approximately 200 Hz. The entire test setup was very rigid.

III. Analysis

A. Nozzle flow characteristics

Internal flow separation from the nozzle wall was studied using measured thrust, wall static pressures, and shadowgraphs. The results of these three methods are shown in Figs. 5, 6, and 7, respectively. The sudden change in nozzle static thrust coefficient (C_V) shown in Fig. 5 at a NPR of about 2.25 indicates a significant flow separation. At lower NPR, the separation region moves upstream and C_V remains high relative to the level predicted with attached flow. With increasing pressure ratio, the separation moves downstream and reaches the nozzle exit around NPR of 3 where measured and predicted levels of C_V are equal. The location of the point of separation in the nozzle can be seen in Fig. 6 by noting the intersection of curves A–F with the flowing full internal pressure distribution curves, G, H. As the nozzle pressure ratio is increased from 1.3 to 3.01, the shock moves downstream through the nozzle. The lower portion of Fig. 6 contains nozzle wall Mach numbers computed from the flowing full pressures (curves G, H) assuming an isentropic expansion.

Figure 7 is a series of shadowgraphs which depict the progression of the shock system with increasing NPR. Shadowgraphs are sensitive to second derivatives of density and, therefore, only show strong disturbances as shocks; weak disturbances, such as expansion waves, are not visible. At a NPR of 1.9 (see Fig. 7a), the shock, which is located just downstream of the throat, is seen emerging from the nozzle, causing a substantial separated flow. As the NPR is increased to 2.2 (Fig. 7b), the shock sits at a location of 65% of the total length. The shadowgraph shows the separated shock leaving the nozzle and forming into a strong Mach disk. There is still a substantial outer core of separated flow. At a NPR of 2.5 (Fig. 7c), the shock is sitting close to the end of the lip with a small region of separated flow. At a NPR of 3 or greater (Fig. 7d), the nozzle is flowing full. A weak shock, seen in Fig. 7d, emanates from the nozzle walls. The The source of this shock may be due to the nozzle shape as noted previously,[7,8] or to slight contour irregularities due to fabrication. In addition, a slight pressure rise is also evident in the nozzle wall pressure distribution data.



INTERNAL
TAP LOCATIONS

Tap No.	X	θ°
40	3.56	0°
41	3.75	0°
42	3.95	0°
43	4.15	0°
44	3.56	180°
45	3.75	180°
46	3.95	180°
47	4.15	180°
48	3.50	270°
49	3.65	270°
50	3.85	270°
51	4.05	270°

Fig. 2 Nozzle geometry and pressure tap location.

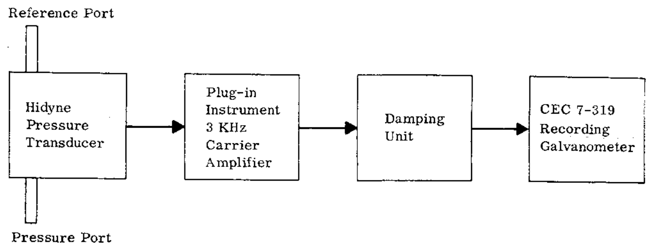


Fig. 3 Instrumentation schematic.

B. Transients

Results of the transducer measurements indicated two dominant frequency ranges: 20–40 Hz and 100–200 Hz. At the lowest nozzle pressure ratio (1.3), all of the pressure fluctuations occurred in the higher frequency range. As the pressure ratio is increased, supersonic flow and separated

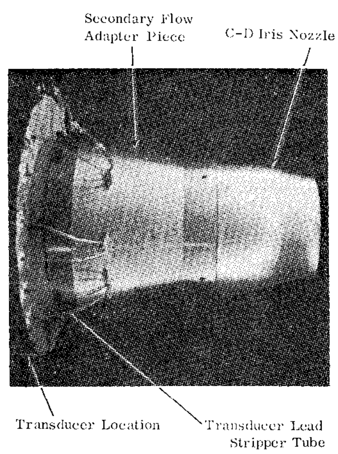


Fig. 4 High response test setup.

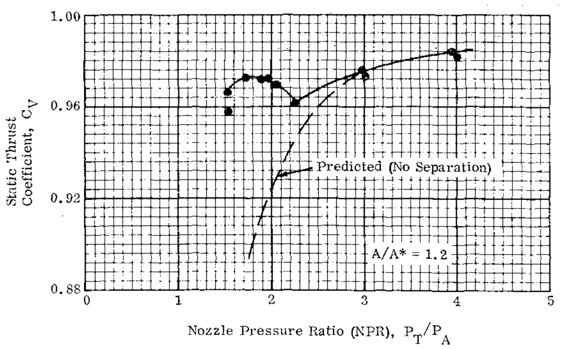


Fig. 5 C-D iris nozzle static thrust.

regions are established in the nozzle. As noted in Fig. 8, at NPRs of 1.75 and 2.0 some lower-frequency oscillations were observed at axial locations near and downstream from the separation point. In general, the magnitude of the pressure fluctuations was within 3.5% of the total pressure. This is small by comparison with the differences in pressure between separated and flowing full pressures which, for example, would be 20% of the total pressure at a NPR of 2.0 (Fig. 6) and $X/L = 0.5$. It was thus concluded that at all NPRs there was no significant unsteady motion of the separated flow or shock system.

At present, there are no reliable analyses for predicting the stability of this type of transonic flow, or the magnitude and type of perturbation required to initiate large magnitude instabilities. The experimental work with a forward facing step[9] is basic in that it establishes large pressure fluctuations for Mach 3–4.5 separated flows with a turbulent boundary layer. A schematic of the pressure fluctuations are shown in Fig. 9. Two mechanisms are postulated for this unsteadiness. The possibility of a cavity-type acoustic oscillation of the entire separated region is discounted[9] because of the lack of measured uniformity of oscillations throughout the separated region and evidence of strong low-frequency (less than

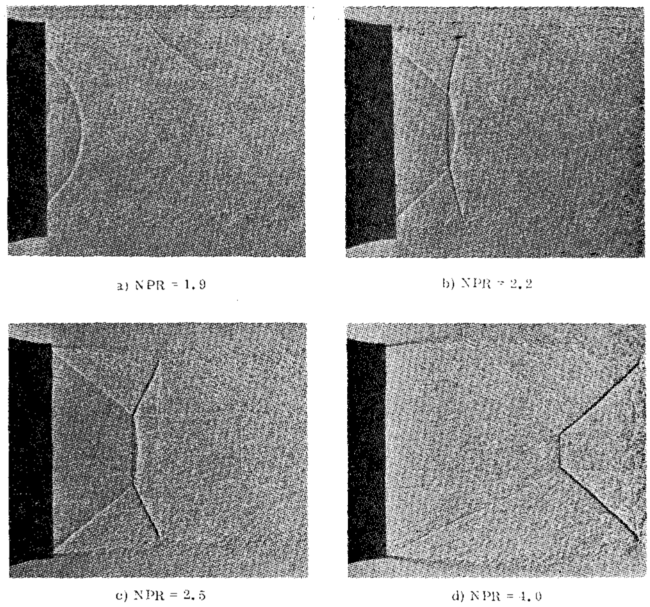


Fig. 7 Shadowgraphs of nozzle jet flowfield ($W_S/W_P = 0$).

1 KHz) energy concentration. A conjecture considered more plausible is that the dividing surface between the shear layer and reversed flow is randomly distorted.

The type of flow studied previously is different from nozzle separation in that the separated flow in nozzles contains a free rather than a bound vortex system (Fig. 9). Additional theoretical studies of the interaction of an oscillating shock with a laminar boundary layer[10] have indicated that, for certain values of shock strength and frequency, self-supported oscillations of the flow pattern occur. It is predicted that at Mach numbers below 1.6 the oscillation is always stable. In our tests, separation within the nozzle occurred at Mach numbers below 1.6 (Fig. 6), so that the latter theory[10] may be

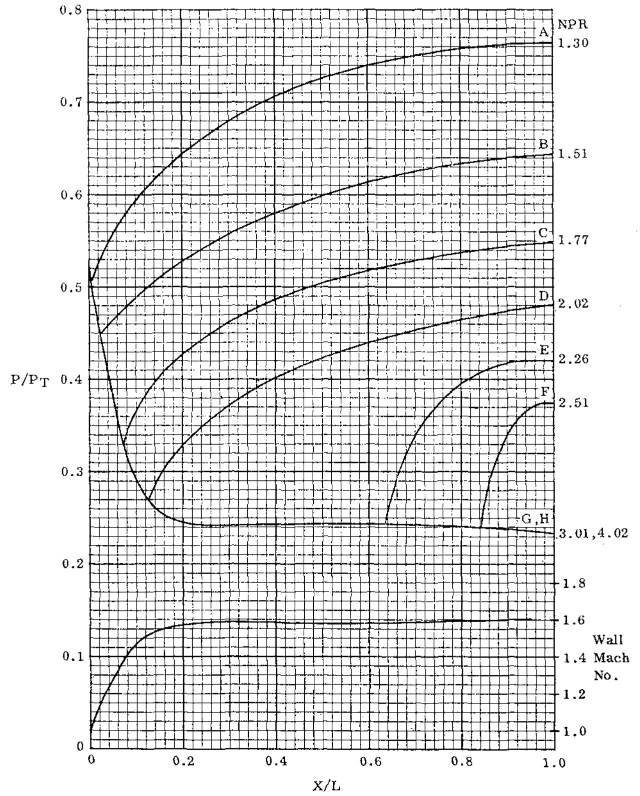


Fig. 6 Internal nozzle pressure distribution and flowing full wall Mach numbers.

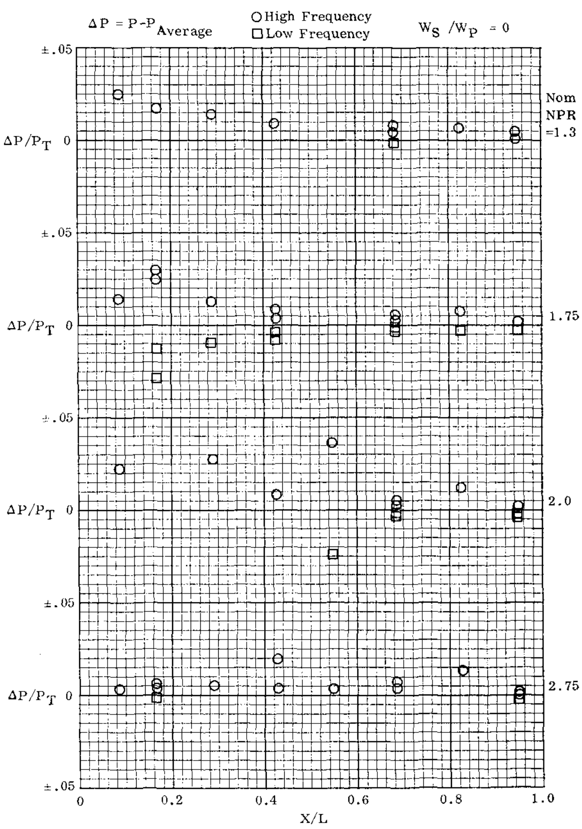


Fig. 8 Pressure fluctuation amplitude.

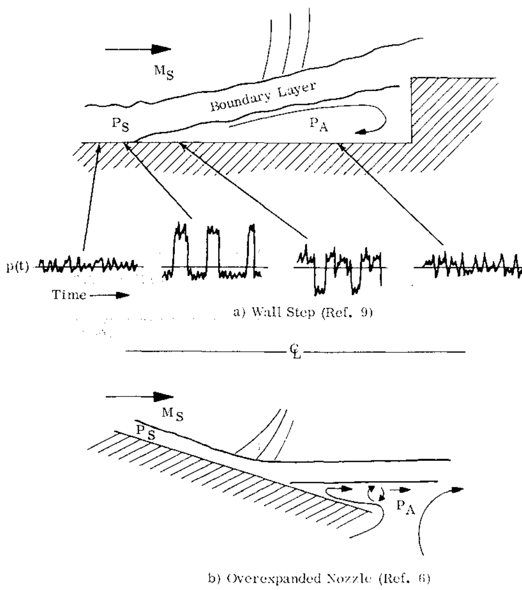


Fig. 9 Schematic of flowfield models.

applicable. It is interesting to note that certain earlier experimental data[2] also indicated no instability for nozzle area ratios less than 1.22.

C. Symmetry

The nozzle exhaust was examined to determine if any asymmetry, where the jet flips from wall to wall, existed in the flow. Visual observation revealed a steady jet. An examination of the shadowgraphs shows an equal separated region on the top and bottom of the nozzle. Also the wall pressure data were the same at 0° and 180°, thus, showing that no asymmetries exist.

D. Hysteresis

It was questionable whether a rigid model, such as was used in the test program, would display the unstable characteristics which might exist for a flexible structure. However, it was noted[4] that if hysteresis occurred in a rigid model, this would indicate an unstable separation phenomenon for a similar elastic configuration. Runs were made with first increasing and then decreasing NPR to determine if hysteresis existed. Pressure leads were connected to the manometer boards to obtain an accurate reading. The test was run with 2% secondary flow to simulate the real flow system more closely, and the results are shown in Fig. 10. A detailed plot of an individual tap is given in Fig. 11. It is seen that hysteresis was not present. From this latter type of plot, separation NPR was determined for a given tap.

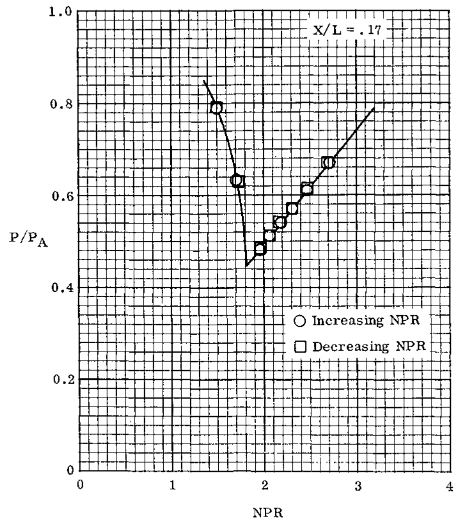


Fig. 11 Hysteresis test—determination of separation for a specific pressure tap.

E. Secondary flow

Since secondary flow is used in aircraft for cooling purposes, its effect on the internal nozzle flowfield was investigated. No appreciable instabilities were noted with the secondary flow. The pressure fluctuations are comparable to the test setup

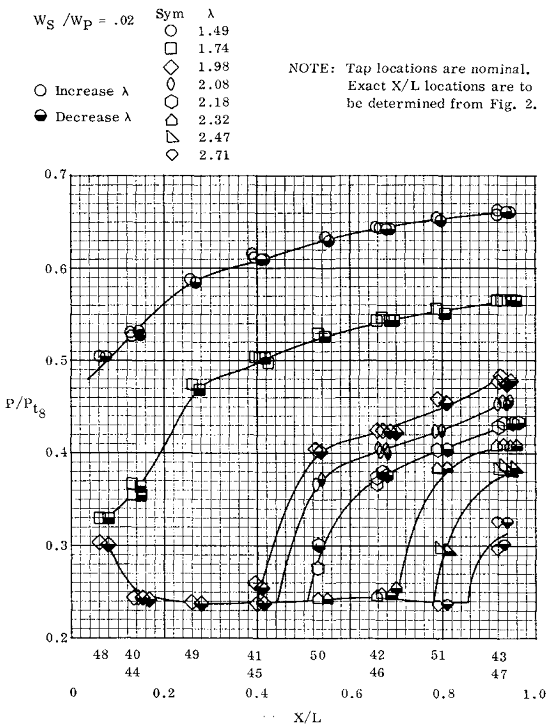


Fig. 10 Hysteresis tests.

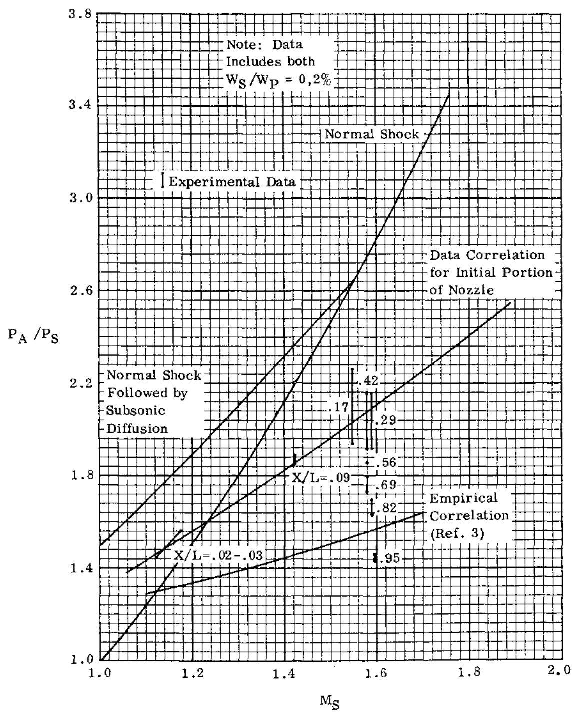


Fig. 12 Separation pressure rise.

without the secondary flow for the high-frequency oscillations; the secondary flow seems to have some damping effect on the low-frequency oscillations. Previous test results[4] also indicated that small secondary airflows had little or no effect on flow instability. The secondary flow appears to have moved the separation shock system slightly upstream.

F. *Over-all separation pressure rise*

Figure 12 shows the measured over-all separation pressure rise at each of the tap locations vs separation Mach number. Included in the figure is the correlation of Ref. 3:

$$\frac{P_A}{P_S} = \left\{ \frac{1 + [(\gamma - 1)/2]M_S^2}{1 + [(\gamma - 1)/2]M_S^2 \, [1 - (U_S^*/U_S^2]} \right\}^{\gamma/\gamma - 1}$$

where U_S^*/U_S (characteristic velocity ratio) $= 0.6$ and γ is the ratio of specific heats. Included in Fig. 12 is the case for which it is assumed that the pressure rise is equivalent to that of a normal shock. Such correlations are useful for predicting the separation pressure ratio. For example, with the nozzle pressure distribution (P/P_T) known (or computed[7]) the Mach number and pressure at an assumed separation point $(M_S, P_S/P_T)$ can be used to determine the over-all pressure rise (P_A/P_S in Fig. 12) and the nozzle pressure ratio for separation (P_T/P_A) computed from $(P_T/P_S)(P_S/P_A)$.

The data in Fig. 12 indicate that two different separation regimes exist, as noted previously.[6] In the first case, separation occurs well within the nozzle $(X/L \leq 0.40)$ while in the second case the separation exists near the nozzle exit. Correlations such as given in Ref. 3 or those obtained from wall steps[7] are only valid for separation well within the nozzle. A much lower-pressure rise is associated with separation near the nozzle exit. For the nozzles under consideration, there is no unique Mach number correlation for more than 50% of the nozzle length. The need for additional theoretical work in this area is thus quite evident.

For separation well within the nozzle, a over-all pressure rise close to an equivalent normal shock value is observed (Fig. 12). This in no way indicates that the classical model of an overexpanded supersonic nozzle[11] is obtained. In the latter model, a normal shock is followed by subsonic diffusion and the flow is assumed always to completely fill the nozzle walls. The results of such a calculation are also shown in Fig. 12; where in this case M_S is assumed to be the normal shock Mach number and one-dimensional flow calculations[11] are used.

Conclusions

The following conclusions are based on testing over a complete range of nozzle pressure ratios: 1) no basic flow instability was noted in a 1.2 expansion ratio convergent-divergent nozzle, 2) the maximum amplitude of pressure fluctuation was greatest near the separation point, but was only 3.5% of the nozzle total pressure, 3) there is no Mach number separation pressure rise correlation for more than 50% of the nozzle flowfield, and 4) additional theoretical work on the stability and steady-state separation characteristics of low area ratio aircraft nozzles is needed.

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