NASA TECHNICAL MEMORANDUM

NASA TM X-52447

NASA TM X-52447

VARIABLE GEOMETRY REQUIREMENTS IN INLETS AND EXHAUST NOZZLES FOR HIGH MACH NUMBER APPLICATIONS

by Milton A. Beheim and Aaron S. Boksenbom
Lewis Research Center
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TECHNICAL PAPER proposed for presentation
at Sixth Congress of the International
Council of the Aeronautical Sciences
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION · WASHINGTON, D.C. · 1968

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INTRODUCTION

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For supersonic aircraft the inlet and exhaust nozzle are critical components of the propulsion system. At supersonic speeds their internal pressure forces produce a large portion of the propulsive force. However, the designs that are most efficient at supersonic speeds perform so poorly at subsonic speeds that intolerable pressure losses and high drag forces exist. As a consequence the geometry of both the inlet and nozzle must be altered as flight speed is varied in order that high efficiency is obtained at both supersonic and subsonic speeds. Although the geometry variations are fairly straightforward from ar aerodynamic viewpoint, the mechanisms that produce these variations and their control systems are not. As a result the design of these components becomes a judicious combination of the aerodynamic objectives and of the practical limitations of aircraft mechanical and control systems. As a part of the renewed effort on airbreathing propulsion which is underway at the Lewis Research Center, particular emphasis is being placed on these problems of making the inlet and nozzle into suitable components of a complete propulsion system rather than limiting the effort to an aerodynamic study of them as isolated components. This effort is integrated with research programs on the other engine components and makes maximum utilization of the propulsion test capabilities of the Center's 10x10 and 8x6 Foot Supersonic Wind Tunnels. At transonic speeds where wind tunnel testing is most difficult because of model size limitations, a flight test program has been initiated to supplement the ground test facilities in order that the full range of critical flight speeds can be adequately investigated.

Many factors influence the basic design of the inlet and nozzle, and greater variety in their shape and form exist than for other propulsion system components. The principal factor is the aircraft mission. It determines the airframe design, the engine cycle, the manner in which the airframe and propulsion system are integrated, the supersonic design Mach number, and the tradeoffs in the relative importance of supersonic and of subsonic performance. In the Center's current effort particular emphasis is being placed on podded engine concepts thereby permitting us to minimize to some extent the range of possible geometric variables. A particularly interesting application of podded engines is to mount the nacelle near the aft lower surface of the wing to shield the inlet from adverse angle of attack effects. By proper shaping of the nacelle and adjacent wing surfaces, favorable interference effects may also be generated which minimize the total aircraft drag[1]. The supersonic design Mach number for our research propulsion systems is in the 2.5 to 3 range, and equal emphasis is placed on obtaining high performance at both supersonic and subsonic speeds. A range of engine cycle characteristics appropriate for both turbojet and turbofan engines is assumed. In the initial testing axisymmetric configurations were selected, but two-dimensional designs are also being prepared. The division of external and internal inlet compression is varied, and both divergent ejectors and plug nozzles are included. In all cases major emphasis is placed on evaluating the relative advantages and disadvantages of the various variable geometry concepts for achieving high propulsion system performance over a broad range of flight speeds. The present status of the effort will be discussed; however, results are incomplete and therefore conclusions are tentative.

INLET SYSTEMS

Inlets which are limited to maximum speeds of about Mach 2 or less generally utilize all-external supersonic compression. At higher speeds to Mach 2.5, all-external compression can still be utilized to obtain high pressure recovery, but the compressive turning of the centerbody becomes so high that steep cowl angles and hence high drag forces result. A Mach 2.5 external compression inlet is shown in Fig. 1. The isentropic spike turns the flow 36°, and the external cowl angle is 25°. In general all inlets in this high speed range require centerbody and/or cowling variations that increase the throat area at speeds less than design so that sufficient air can be provided to the engine. With all-external compression inlets this centerbody variation may be feasible with a two-dimensional design, but in an axisymmetric design the mechanical system problems of collapsing a multiple curved leaf spike are formidable. At design speeds above Mach 2.5 these problems, of course, persist; but in addition even the pressure recovery begins to deteriorate because of the compressive turning limitations of the shock structure which are discussed in Ref. 2. Shielding techniques to reduce cowl drag have been attempted with only limited success[3]. Therefore to attain high levels of performance at speeds above Mach 2, mixed compression inlets become essential. Since the supersonic compressive turning is now reflected between the centerbody and the cowling inner surfaces, the centerbody and the external cowl angles are considerably less. An inlet of this type, with 40 percent of the supersonic area contraction occurring externally, is also shown in Fig. 1. The internal performance of both these inlets at their design speed and at zero angle of attack is shown in Fig. 2. Although the mixed compression inlet had a somewhat higher performance level, considerably more effort was expended on development of its boundary layer bleeds at shock impingement points. Not apparent on the figure is the fact that the external compression cowl drag was 8 times that of the other inlet. Distortion characteristics of both inlets were similar; however, stability margin was somewhat greater with the mixed compression design. This stability in both

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inlets was not accomplished by normal shock spillage over the cowling but rather was a result of increased flow through the boundary layer bleed systems as the normal shock entered the throat region. Hence, it was sensitive to bleed system design. Ideally the stability margin of the inlet in combination with its control system should be adequate to absorb most of the transient disturbances that occur in a propulsion system without requiring supercritical inlet operation during steady operation prior to the transient. Obviously this condition cannot generally be provided for large transients such as a compressor stall, and it may be difficult to accommodate a multiplicity of smaller disturbances occurring simultaneously.

If the inlet flow is reduced below the minimum stable value, large transients occur: the external compression inlet undergoes large amplitude buzz, and the mixed compression inlet has a hard unstart followed by low amplitude buzz. Hence the control system requirements to avoid these undesirable regions of inlet operation are equally demanding for both inlets. The external compression inlet could be restarted more rapidly since the centerbody geometry does not require adjustment; however, the adverse effects of hard buzz on engine operation may be just as serious as is the unstart effect of a mixed compression inlet. Hence, in this regard there does not appear to be a clear cut advantage of one inlet versus the other. The mixed compression inlet performance during an unstart and restart cycle is shown in Fig. 3. In this particular case the inlet is restarted by centerbody translation. The initial large transient in pressure and flow generally causes compressor stall. To aid in quick recovery of propulsion system operation, the restart procedure should be rapid and properly coordinated so that engine operating problems are not aggravated. Immediately after an unstart, low amplitude buzz is attenuated by opening a bypass valve at the end of the subsonic diffuser to choke the throat. Centerbody geometry is varied to open the throat so that the normal shock can be readmitted, but in doing so the bypass is properly controlled so that engine face distortion is minimized by keeping the inlet near peak recovery at each centerbody position.

The centerbody variation required to effect the restart is reduced considerably by a favorable normal shock-boundary layer interaction of the type described in Ref. 4. A prediction of the inlet internal area ratio permitting restart is required to define its variable geometry requirements. Figure 4 presents restart data in terms of the internal area ratio and average cowl lip Mach number. Data from prior tests of axisymmetric inlets are shown as lines which are marked with the inlet design bleed flow. For comparison, curves are also presented for the isentropic area ratio and for the normal shock Kantrowitz contraction ratio. Although the upper curve has been generally used as the criteria for inlet restart, it was known to be conservative. A sketch on the figure indicates the flow field that was measured on a Mach 3 mixed compression inlet just prior to restart[4]. A separated region was evident, and the separation angle and the associated oblique shock pressure rise was determined to be in agreement with the minimum value required to separate a turbulent boundary layer at the cone surface Mach number. It was also determined that the flow area above the separated region was choked at the throat, and at the cowl lip station it was related to the throat area by the isentropic area relationships. Therefore the centerbody position could be computed which would permit the entering flow to pass through the throat without normal shock spillage. The predicted and measured restart geometries of both the Mach 3 and the Mach 2.5 inlets are indicated by the data points and were substantially in agreement with the curves for other inlets. This analysis of the flow field thus permits accurate predictions of the restart geometry requirements.

Inlet unstart can also be triggered by excessive angle of incidence effects. The tolerance of the inlet is sensitive to normal shock position and to the design of bleed systems in the supersonic portion of the diffuser. The effect of angle of incidence on inlet operation with the normal shock just aft of the throat region is shown in Fig. 5 for various centerbody positions. Ideally the tolerance should be adequate at the design position to accommodate sudden transients such as gusts. Additional tolerance to accommodate aircraft maneuvers can then be provided by proper control of centerbody position. The angle of incidence effects on the external compression inlet are also shown on the figure. Distortion is one of the critical inlet parameters for operation at angle of incidence since compressore stall must be avoided during the maneuver. Lower distortions were achieved in the high angle of incidence range with the mixed compression inlet than with the external compression concept. For some aircraft even greater tolerance to angle of incidence is desired. With inlets of the type shown here, adequate shielding by airframe surfaces would then be required.

It is apparent that careful control of normal shock position is required to avoid regions of inlet instability such as unstart and buzz. This is normally done by a closed loop control of the inlet bypass which senses terminal shock position. The design of this control system is influenced by the dynamic response characteristics of the terminal shock position to upstream and downstream disturbances. Recent results from Ref. 5 are presented in Fig. 6 to show the open loop shock response to a downstream sinusoidal disturbance at various frequencies. The inlet was terminated by three different geometries: a large pipe with a downstream choke point, a choke point at the simulated compressor face, and an actual turbojet engine. It is apparent that the engine effect was adequately simulated by the compressor face choke point. At low frequencies below 25 Hz the amplitude and phase shift of the response was typical of that predicted by first order lumped parameter analysis, but at higher frequencies resonances occurred which are typical of a distributed parameter system. These resonances substantially increased the shock sensitivity to high frequency disturbances. The resonant frequency was relatively insensitive to the termination geometry but would be expected to vary inversely with model scale. A one-dimensional mathematical analysis of shock position dynamics has been developed[5] which corroborates these experimental results. It is based on a linearized set of equations across the terminal shock combined with linearized wave equations for the subsonic duct. These equations utilize flow, total pressure, and entropy as the dependent system variables and are far simpler in application than other accurate procedures such as the method of characteristics. To obtain frequency response the method provides a closed form solution which is solved by straightforward matrix methods. The complete set of equations can also be easily used on an analog computer for synthesis of propulsion system dynamics. For inlets with simple throat bleed systems this analysis provides excellent correlation with experiment. However, with complex bleed systems such as those used in the particular mixed compression inlet shown here, empirical procedures are required to define effective flow area distribution in the vicinity of the normal shock wave and to define bleed system flow coefficients.

Shock dynamic response to an external disturbance was also obtained with the test apparatus pictured in Fig. 7. A large plate was oscillated in the pitch plane to generate sinusoidal variations in both angle of incidence and local stream Mach number. Results shown in Fig. 8 for the Mach 2.5 mixed compression inlet discussed in earlier figures indicate greater attenuation and phase lag as frequency increased. However, also shown on this figure is the response for a different mixed compression inlet at Mach 3 which showed greater sensitivity of shock motion to this disturbance as its frequency increased. Causes for these differences in shock response characteristics have not yet been determined.

A research program directed to the control of the variable bypass has been completed recently and some results are discussed in Ref. 6. The control concept used is illustrated in Fig. 9. Two static pressure sensors are required: one indicates shock position and the other provides anticipation of disturbances originating from the engine. The bypass actuators were substantially faster than those used in normal flight systems and provided flat response to 100 Hz. The closed loop performance of this control is compared to the open loop response in Fig. 10. The gains of the control system were limited such that the shock response did not excessively exceed the open loop response at the inlet-engine system resonant frequency. The large attenuation of shock motion at the lower frequencies is evident. In a similar manner the effectiveness of this bypass control in minimizing shock response to the external disturbance is shown in Fig. 11.

Additional control capabilities are required in the event of an inlet unstart. Fig. 12 shows a schematic representation of the restart control which was investigated. The ratio of an internal cowl lip static to throat total pressure was used to detect unstart. Proper coordination of centerbody position and of bypass opening were required to provide the best flow conditions to the engine as possible under the circumstances. A typical unstart transient followed by a controlled restart cycle is shown in Fig. 13. The arrows indicate increasing values of the variables. The total restart cycle took 1.46 seconds and was limited by the centerbody actuator translational rates.

In the process of varying inlet geometry, another range of inlet operation that must be avoided is that which produces high levels of distortion at the entrance to the compressor. This occurs if the inlet normal shock is in an excessively supercritical position or if the centerbody geometry variations are such that the supersonic throat Mach number just ahead of the normal shock is excessively high. In either case strong shock-boundary layer interactions occur which disrupt normal operation of the subsonic diffuser. Typical steady state total pressure contours at the compressor face of the mixed compression inlet are shown in Fig. 14. The centerbody support struts divided the flow passage into three ducts at the station where these measurements were made. Also shown on the figure is an individual total pressure trace as a function of time. It shows that the distortion of the air flow has a time dependency which can be quite large and has been termed "dynamic distortion." Adverse effects of distortion on engine operation have long been recognized. Typical effects include: deterioration of compressor stall margin, high levels of stress in compressor blades and discs, and adverse temperature gradients in the burner. The first two of these effects are further aggravated by the dynamic characteristics of distortion but quantitative results are limited.

Effects of boundary layer bleed system design on distortion in the Mach 2.5 mixed compression inlet is shown in Fig. 15. Both steady state distortion and the average root mean square of the dynamic component are shown for varying amounts of supercritical inlet operation produced by changing the bypass opening. Large amounts of bleed reduced distortion as expected during critical inlet operation where the normal shock was near the bleed system. However, as the shock was pulled supercritical, large increases in both steady and dynamic distortion occurred regardless of which bleed system was used. Radial distribution of steady and dynamic distortion are shown in Fig. 16 for critical and supercritical inlet operation as controlled again by bypass opening. At pressure recoveries of 0.911 and 0.770 the steady state pressure profiles were relatively flat, but dynamic distortion increased significantly at the lower recovery. At a recovery of 0.731 the centerbody flow tended to separate, and the dynamic distortion was greater in the high pressure portion of the profile.

In an earlier test of a Mach 3 mixed compression inlet which utilized pressure sensors with a flat response to 200 Hz, a power spectral analysis of the dynamic distortion indicated a flat spectrum at critical and supercritical operation. In recent tests of the Mach 2.5 mixed compression inlet, pressure sensors with a flat response beyond 1000 Hz were used. For most operating conditions of the inlet, prominent frequencies were observed. Typical pressure traces from different sensors at the compressor face station shown in Fig. 17 indicate a prominent frequency of 300 Hz. Various sensors were 180° out of phase at different points within the flow passage between struts. In an adjacent passage the prominent frequency was about 295 Hz; and, as shown by the throat exit static trace, the shock responded at the sum of these frequencies. These results suggest that the normal shock oscillations were coupled with a fairly complex acoustic mode of the duct and hence could be influenced by specific details of each subsonic diffuser design.

An indication of the magnitude of the effect that distortion has on the stall margin of a compressor when operated with the Mach 2.5 mixed compression inlet is shown in Fig. 18. A wide variety of inlet operating conditions were used at Mach 2.5 to obtain these results. Although the steady and dynamic distortion values were not always related in a consistent manner, increases in dynamic distortion were obtained with only small increases in steady distortion. As shown on the figure, deterioration of stall margin was consistent with increased dynamic distortion though not necessarily in a linear manner. It would be anticipated that qualitatively similar effects would be observed in other inlet-engine combinations, but there is insufficient information to generalize the quantitative effects.

The use of vortex generators in a reasonably long and gradual subsonic diffuser has proved to be a highly effective means in controlling steady state distortion. In general they can be designed in such a manner that the discrete vortices they create are dissipated at the compressor face and hence would not aggravate compressor operating problems. Although vortex generators were not required in this Mach 2.5 mixed compression inlet during normal operation, they were very effective in suppressing a tendency of the flow to separate from the centerbody as the bypass was opened. As shown in Fig. 19 they were also very effective in reducing dynamic distortion. Results obtained with and without vortex generators on the centerbody and cowling are presented. Apparently the vortex generators energized the boundary layer and hence stabilized the shock-boundary layer interaction over the entire supercritical operating range of the inlet. Thus, the vortex generators are a powerful tool in desensitizing the inlet to adverse effects of off-design operation.

EXHAUST NOZZLES

The divergent ejector is generally used on aircraft which require high performance at speeds above Mach 2. For high performance at low pressure ratios the divergent shroud expansion ratio must be decreased. In some designs it is done entirely by mechanical means wherein the shroud internal and external surfaces are constructed of multiple flaps which overlap as the exit area is closed. Photographs of a model utilizing this concept are shown in Fig. 20. In this report it will be referred to as the variable flap ejector. In other designs a similar mechanical adjustment of the shroud is used, but in addition auxiliary inlets are provided as an aerodynamic means of reducing the shroud expansion ratio by partially filling it with ambient air. Design details of both these nozzles varies considerably depending upon the application. This is particularly true when thrust reversal and sound suppression devices are also required. In either case there is a desire to be able to vary the movable portions of the shroud and auxiliary inlets between inner and outer stops by means of the internal and external air loads rather than by means of mechanical actuation.

An entirely different concept of nozzle design which has had only limited use is the plug nozzle. Early emphasis was placed on isentropic plug designs wherein the contour of the plug was such that it cancelled the expansion fan originating from the throat flap. Although static performance was excellent, it suffered large performance losses due to external flow effects when operated at low nozzle pressure ratios. This loss originated from the high base drag of the steep throat flap and from the concurrent overexpansion effects on the plug surface which were produced by the low base pressures[7]. A more recent innovation in plug nozzle design[8] is to utilize a low angle conical plug and hence low throat flap angles are permitted (fig. 21). Since this configuration has low performance at high pressure ratios as a result of excessive jet pluming just downstream of the throat, a shroud is then required which provides sufficient internal expansion to reduce the jet pluming effects. Because the internal expansion must be eliminated for operation at low pressure ratios, a variety of variable shroud concepts and mechanisms are possible. The most straightforward concept appears to be a cylindrical shroud which is translated to vary the internal expansion ratio and will be the only one discussed hereafter.

Another variable geometry requirement of the plug nozzle is that throat area be modulated for afterburner use. The amount of modulation varies depending upon engine cycle and can strongly influence the choice of the mechanical system design utilized for this purpose. In concept it can be varied by an iris primary flap, by collapsing a portion of the plug surface, or by relative translation of the plug with respect to the primary flap.

A serious problem in the application of any plug nozzle to an afterburning engine is that of cooling the plug surface and its support structure. Although film cooling with secondary air is used in the ejector nozzles, excessive quantities of high pressure air may be required if applied to plug cooling. An interesting alternative is to utilize some of the heat sink capacity of the afterburner fuel in a regenerative cooling arrangement. Tentatively it appears that the plug nozzle mechanical systems may be simpler than those of ejector nozzles and hence could be of great utility in applications where long life and low maintenance are required. Hence, considerable effort directed to this cooling problem seems justified. The Center is proceeding with more detailed studies of plug cooling by means of both air and fuel.

In recent tests the isolated nozzle performance was obtained with and without external flow with models of both variable flap ejector and plug designs. A comparison of results applicable to a turbojet engine cycle is shown in Fig. 22 over a broad range of flight speeds. This particular ejector nozzle utilized a floating divergent shroud positioned between inner and outer stops by the air loads. The 10° half angle plug nozzle utilized a sliding cylindrical shroud which was positioned so as to maximize aerodynamic performance. For purposes of this figure, secondary airflow requirements were assumed to be equal for both nozzles. The plug was capable of providing performance that equaled or exceeded that of the ejector nozzle at all flight conditions. A part of the ejector nozzle performance loss at low speeds was partly attributed to internal overexpansion of the primary jet flow. The inner stops were selected to provide this overexpansion in order to avoid a potential instability of the floating shroud which may occur if the shroud is allowed to close further.

A significant loss in ejector nozzle performance occurred at subsonic cruise. A large part of this loss is attributed to the boattail pressure drag. This drag force may be a significant fraction of the overall aircraft drag at this flight condition. It is sensitive to the jet exit static pressure ratio and to details of the afterbody shape and of the approaching boundary layer. The effects of boattail shape on the transonic drag rise curve are shown in Fig. 23 and are discussed in more detail in Ref. 9. In this figure the radius at the juncture between the cylindrical nacelle and a 15° conical boattail is varied. It is apparent that increasing the radius delayed the drag rise to higher Mach numbers. The effect of variations in boundary layer thickness just ahead of the boattail is shown in Fig. 24 for two boattail shapes at Mach 0.9. In both instances thicker boundary layers tended to decrease the boattail drag.

The plug nozzle performance is also influenced by external flow with the shroud retracted. Typical results showing the effects of nozzle pressure ratio on thrust efficiency at several free stream Mach numbers are shown in Fig. 25. Quiescent test results are also indicated. The appropriate schedule of pressure ratio with Mach number varies depending upon the specific engine and aircraft design. The rather sharp performance drop at the low pressure ratios which may be appropriate for subsonic cruise is quite evident At high pressure ratios the efficiency also dropped rapidly as free stream Mach number increased. The nature of this latter effect is illustrated in Fig. 26. During quiescent tests the jet boundary was at constant static pressure and incident waves, which originated from the initial jet pluming, reflected as equal strength waves of opposite nature. The periodic variations in plug pressure produced nearly the ideal thrust. However, with external flow the jet boundary was no longer at constant static pressure and incident waves may be only partially reflected but also partially transmitted into the external flow. The initial low pressures still existed on the plug surface, but the succeeding compressions and expansions were attenuated and the plug thrust was less than ideal. Therefore, the initial jet pluming must be carefully controlled by utilizing the proper shroud position and by controlling the flow of secondary cooling air between the jet boundary and the shroud wall.

The effect on nozzle performance of secondary flow between the jet and the shroud is shown in Fig. 27. With the shroud extended, relatively small amounts of flow produced significant gains in performance. Cooling requirements of the shroud may dictate greater amounts of flow than is required for the performance gain. Even with the shroud retracted, secondary flow through the gap at the boattail shoulder improved performance. Larger amounts of flow were desirable than were required with the shroud extended. Apparently this flow improved performance of the retracted shroud configuration by increasing the boattail pressures.

Nozzle cooling is normally accomplished with films of secondary air. Movable components, such as the sliding shroud, will undoubtedly continue to require this cooling technique. A variety of empirical film cooling correlations are available in the literature. One that appears to be most appropriate for aircraft nozzles is that of Ref. 10 which was originally developed for film cooled convergent-divergent rocket nozzles. A comparison of measured and predicted wall temperature distributions for a cylindrical shroud on an afterburning turbojet engine are shown in Fig. 28. A straightforward application of the correlation overestimated the wall temperature, but by applying radiant heat transfer corrections, good agreement with experiment was achieved.

To minimize the heat load on the plug surface cooling system and to minimize the exhaust system length, plug truncation appears desirable. The resulting effect on performance is shown in Fig. 29. The effect was more noticeable at a low pressure ratio than at a high value but, nevertheless, at least 25 percent of the plug length can be removed without producing significant performance losses. If greater truncations are used, bleed flow in the base of the plug tends to minimize the loss in performance.

Virtually all variable divergent shrouds have been subjected to aerostability problems in their closed positions resulting either from oscillations in the location of the sonic point or from aeroelastic oscillations of the shroud structure triggered by overexpansion of the internal flow(11). The plug nozzle is no exception in that it too may be subject to aeroelastic instabilities produced by oscillations in plug pressures coupled with the elasticity of the plug support system. As indicated on Fig. 30 a cantilevered plug support may be used instead of a strut support in order to minimize the heat load on the support cooling system. A displacement of the plug from its neutral position will cause an unsymmetrical air load on the plug and a self-exciting instability can result. As indicated on the figure, instabilities were noted both with the shroud extended and retracted. When retracted the instability occurred with the truncated plug at the highest pressure ratio tested with a free stream Mach number of 0.85, but it was not present with a full length plug. With the shroud extended, instabilities were observed when the jet was at abnormally low pressure ratios. A moderate instability occurred at a pressure ratio of 2 both quiescently and at Mach 0.85. These instabilities apparently were a result of overexpansion of the flow between the plug and shroud surfaces. A very severe instability occurred at Mach 0.85 with the jet off. These results indicate that plug stability can be influenced by external flow effects and therefore wind tunnel tests may be required to establish the required stiffness of the plug support structure.

With the shroud retracted for operation at subsonic speeds, the plug nozzle performance is sensitive to the relative size of the plug and its boattail area. Although a systematic variation of these variables has not yet been completed, some results such as those of Fig. 31 are available. For a given throat to nacelle area ratio, plug size was increased and, concurrently, boattail area decreased. At this particular free stream Mach number and nozzle pressure ratio, highest performance was obtained with the largest plug. For reference purposes the variable flap ejector performance is also shown (for identical flow conditions) as an extreme case wherein the plug size was reduced to zero. Effects of varying the boattail juncture radius, R, are indicated.

A comparison of nozzle performance with an iris primary and with a translating primary flap for throat area control is presented in Fig. 32. These results correspond to acceleration power settings with full afterburning. When the shroud was retracted the performances were nearly identical. With the shroud extended to a position relative to the plug which was equal for both primary nozzles, the translating primary performance was somewhat less because of greater divergence losses at the shroud discharge.

Results discussed thus far were obtained with isolated nozzle models. Significant changes may occur, however, as a result of airframe installation effects[1]. The installation may produce circumferential variations in boundary layer characteristics, in pressure and velocity distributions, and in flow directionality of the external flow. As indicated earlier a type of installation which is of general interest is a podded engine mounted near the aft lower surface of the wing such as that illustrated in Fig. 33. To provide transonic flight test data on complete propulsion systems of this type the Center has modified an F106B aircraft so that research nacelles can be slung beneath the wing in the manner indicated. Major changes may be made in both inlet and nozzle design just as is done in wind tunnel testing. Because this flight test facility permits use of larger scale propulsion system and airframe components than is possible in existing transonic wind tunnels, more extensive instrumentation and hence greater accuracy in evaluating inlet and nozzle components is achieved. In addition to pressure and temperature instrumentation, the nacelles are suspended by hinged links thus permitting a direct measurement of the net propulsive force. Operation of the modified aircraft was initiated recently.

Wind tunnel model tests are also being conducted to provide correlation between flight test data and the various tunnel test techniques. Recent results showing installation effects on a variable flap ejector nozzle are shown in Fig. 34. The isolated boattail pressure drag showed the characteristic transonic drag rise. However, when installed on the F106 research nacelle, the small scale wind tunnel model results indicated a very substantial favorable interference drag effect particularly at speeds of interest for subsonic cruise. By comparing isolated and installed nacelle pressure distributions (Fig. 35) it is apparent that flow spillage effects at the front of the nacelle were amplified by reflection from the wing lower surface producing a strong terminal shock which influenced the boattail pressure distributions. As shown in Fig. 36, increasing free stream Mach number caused the terminal shock to progress further down the nacelle, and the abrupt rise in boattail drag at Mach numbers greater than 0.95 occurred when the terminal shock moved downstream of the boattail. These results suggest that the interference effect would be sensitive to details of inlet flow spillage and also to nacelle shape and fineness ratio.

CONCLUDING REMARKS

Aircraft designed for efficient operation at speeds in excess of Mach 2 will require sophisticated inlet and exhaust systems with variable geometry features in order to achieve good performance over the entire airplane flight spectrum. A realistic compromise between performance and complexity requires a thorough understanding of the inlet and exhaust systems and the interaction between these components and other elements of the propulsion system and airplane. Current programs at the Lewis Research Center are contributing to this understanding through analytical and experimental work using the supersonic wind tunnels and the F106 flight research configuration. This research, still in its early phases, has already revealed the importance of several technical areas. High frequency dynamic phenomena have been identified in inlets and the importance of interaction between inlet and engine in the presence of these phenomena has been demonstrated. The feasibility of high response control of inlet shock position has been demonstrated in a high performance mixed compression inlet. Experimental programs have established the efficacy of throat bleed and vortex generators in enhancing the practicality of high performance inlets. In the exhaust systems area, good supersonic design point performance can be obtained with several different designs. However, details of geometry variation differ and complication is extreme in some areas. Low angle plug nozzles have shown considerable potential in offering a sound compromise between performance and complexity. Additional nozzle results have demonstrated that boattail drag has an important effect on transonic performance. Boattail drag can be reduced by proper attention to the details of external geometry, by careful installation on the airplane, and possibly by the introduction of plug nozzles. Research in these areas is continuing.

SYMBOLS

A	area
AB	afterburner
CB	inlet centerbody
C_{D_β}	boattail pressure drag coefficient
C_p	static pressure coefficient
D	drag
D_b	base diameter
D_i	inlet cowl lip diameter
D_M	model nacelle diameter
F	gross thrust
K	gain
L	plug length
M	Mach number
m	mass flow rate

P	total pressure; time averaged value for steady-state data, instantaneous value for dynamic data
ΔP	difference between maximum and minimum steady state total pressures
ΔP_{rms}	root mean square of fluctuating total pressure component
P.R.	pressure ratio
p	static pressure
S	Laplace operator
w_c	corrected flow
x	axial distance or axial centerbody translation from design position
δ	boundary layer thickness
$\omega\sqrt{\tau}$	corrected secondary flow ratio

Subscripts:

b	bypass
CB	centerbody
c	control
com	command
cowl	cowl lip station
i	ideal
L	local
max	maximum
min	minimum
op	operating point
p	primary
s	secondary
0	free stream condition
1	inlet throat region
2	compressor face
7	primary jet

Superscripts:

$-$	average value at a station
*	sonic

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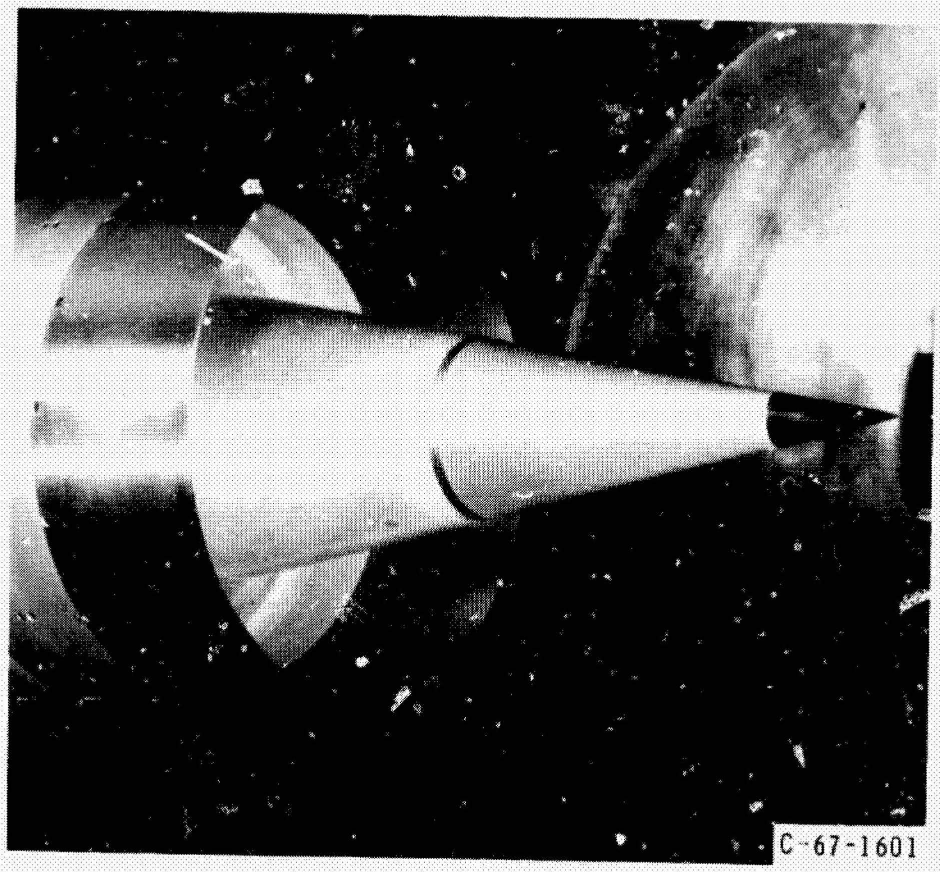
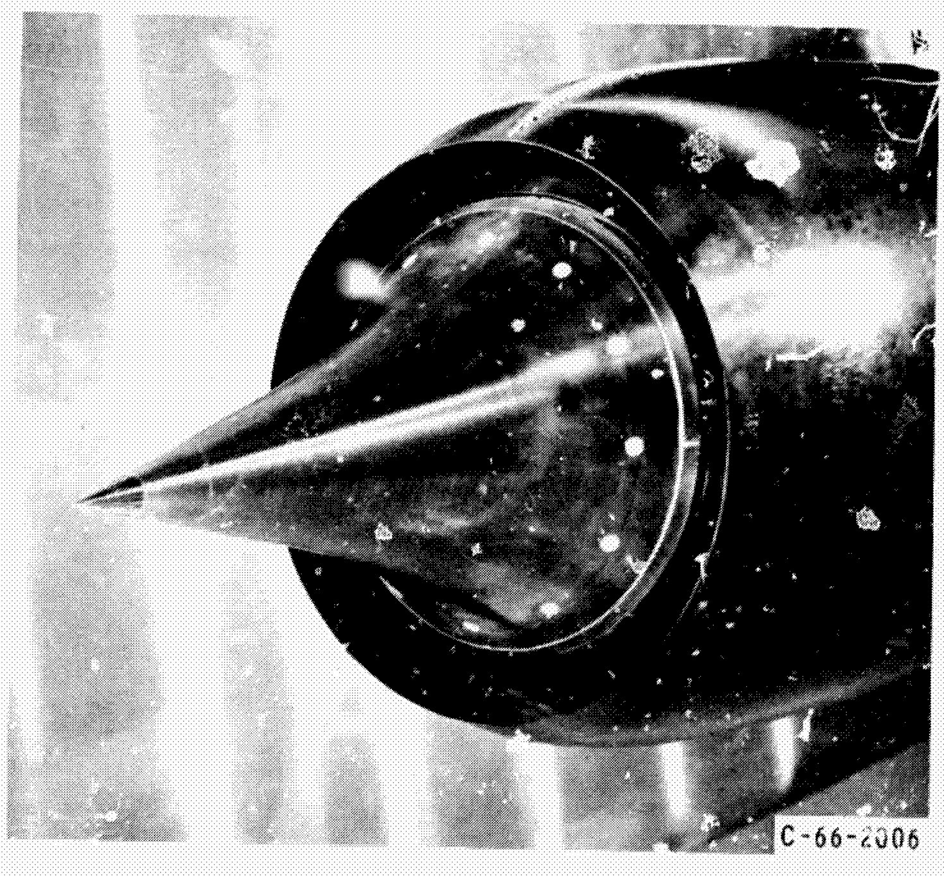


MIXED COMPRESSION INLET



ALL EXTERNAL COMPRESSION INLET

FIGURE 1. - MACH 2.5 INLETS.

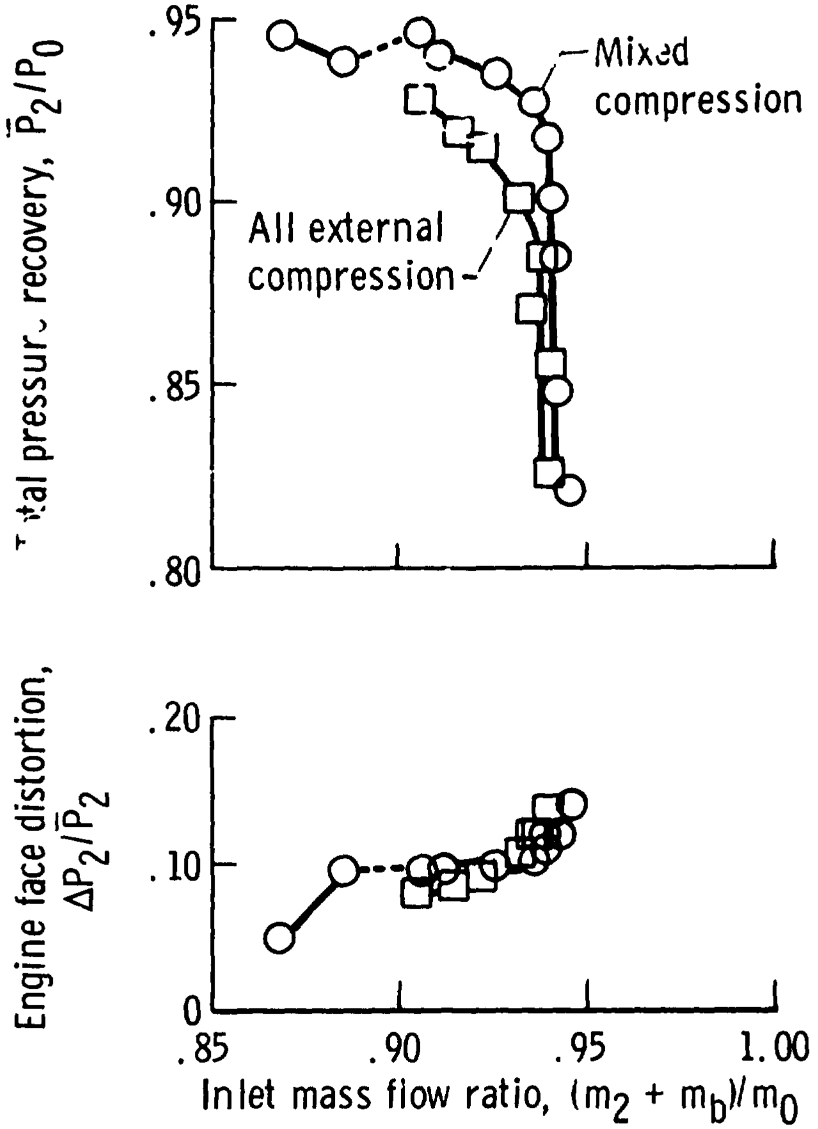


Figure 2. - Experimental Mach 2.5 inlet performance.

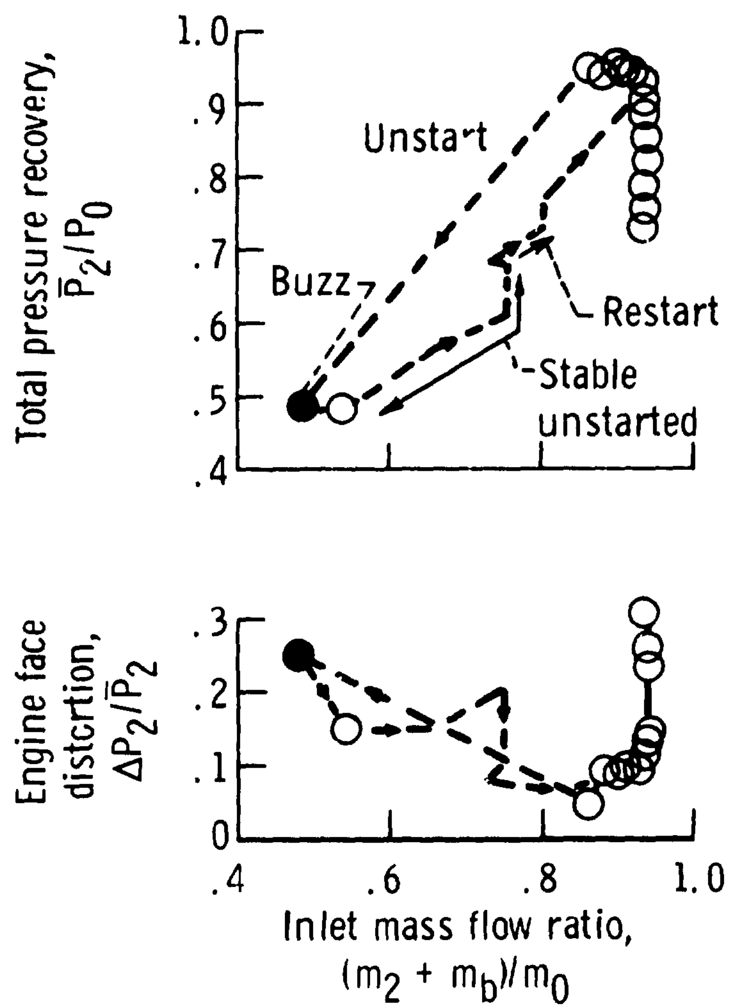


Figure 3. - Mixed compression inlet restart performance.

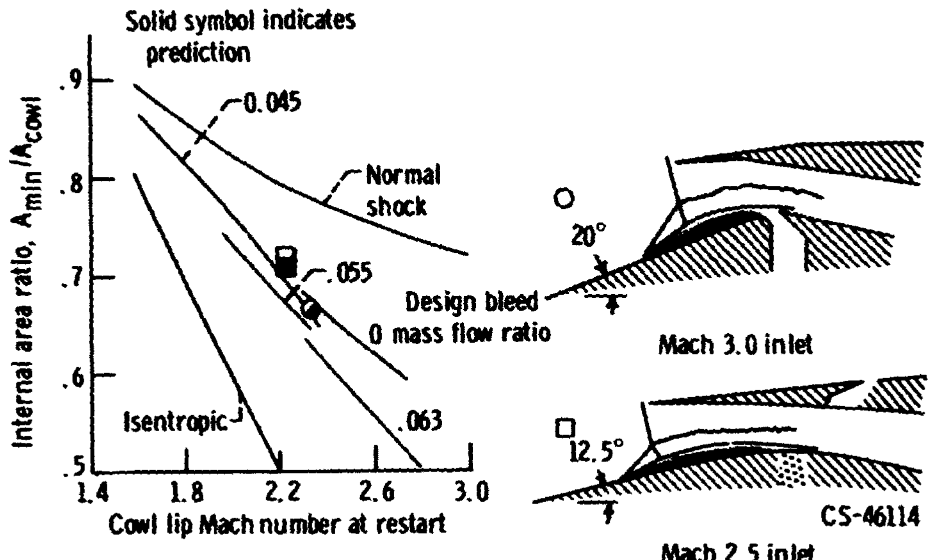


Figure 4. - Inlet restart prediction.

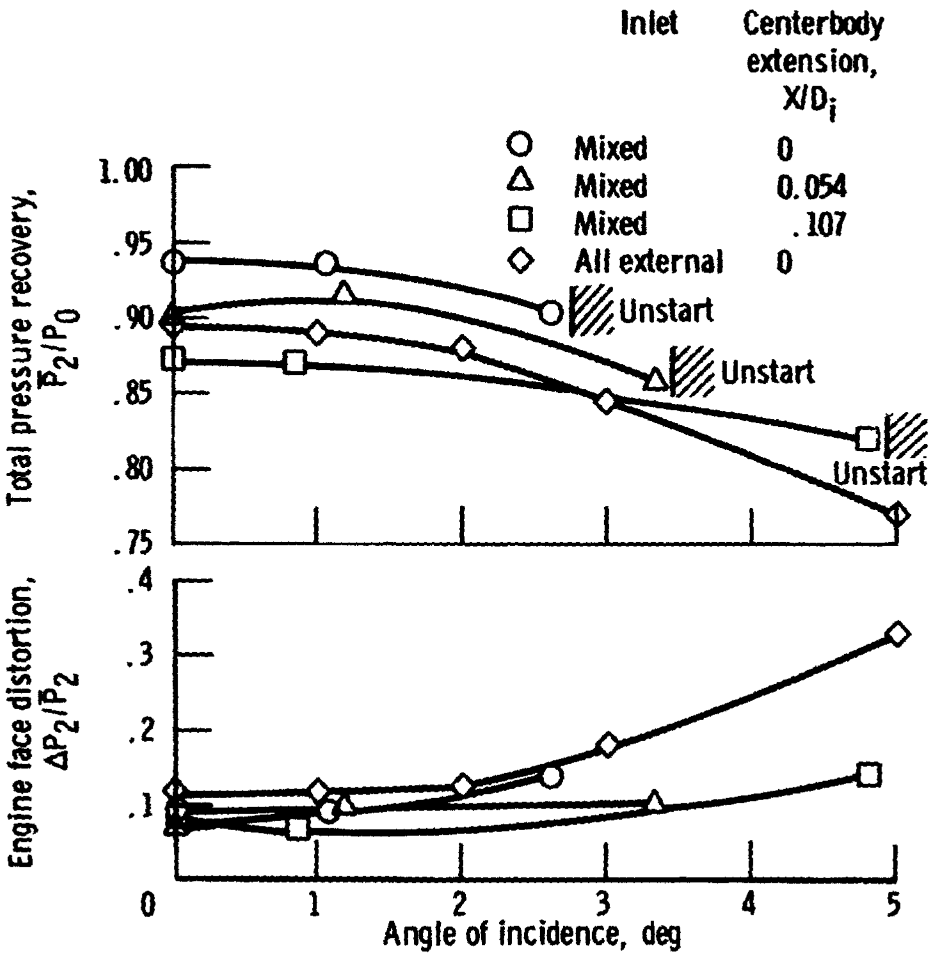


Figure 5. - Angle of incidence effects on peak inlet performance at Mach 2.5.

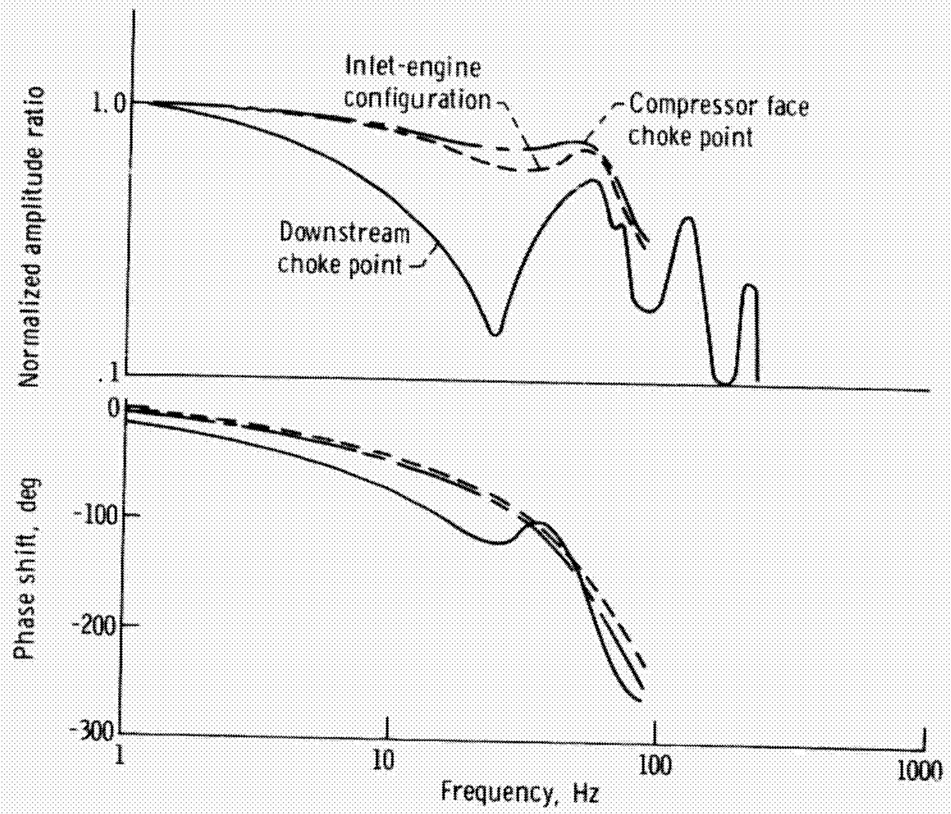


Figure 6. - Dynamic response of shock position to internal disturbance.

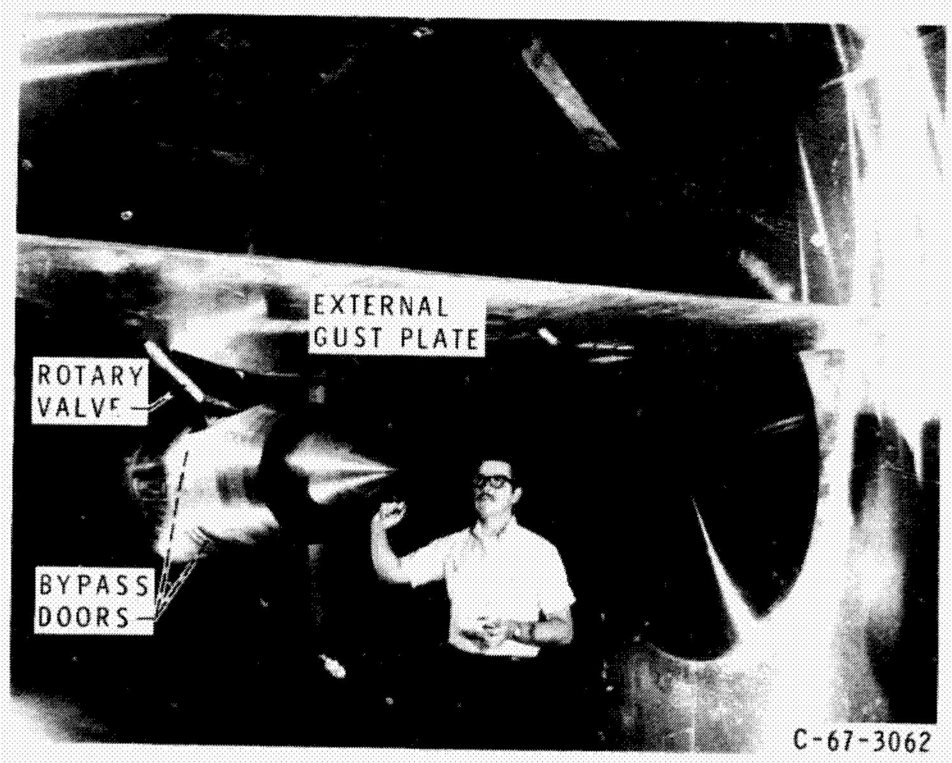


FIGURE 7. - PHOTOGRAPH OF INLET MODEL.

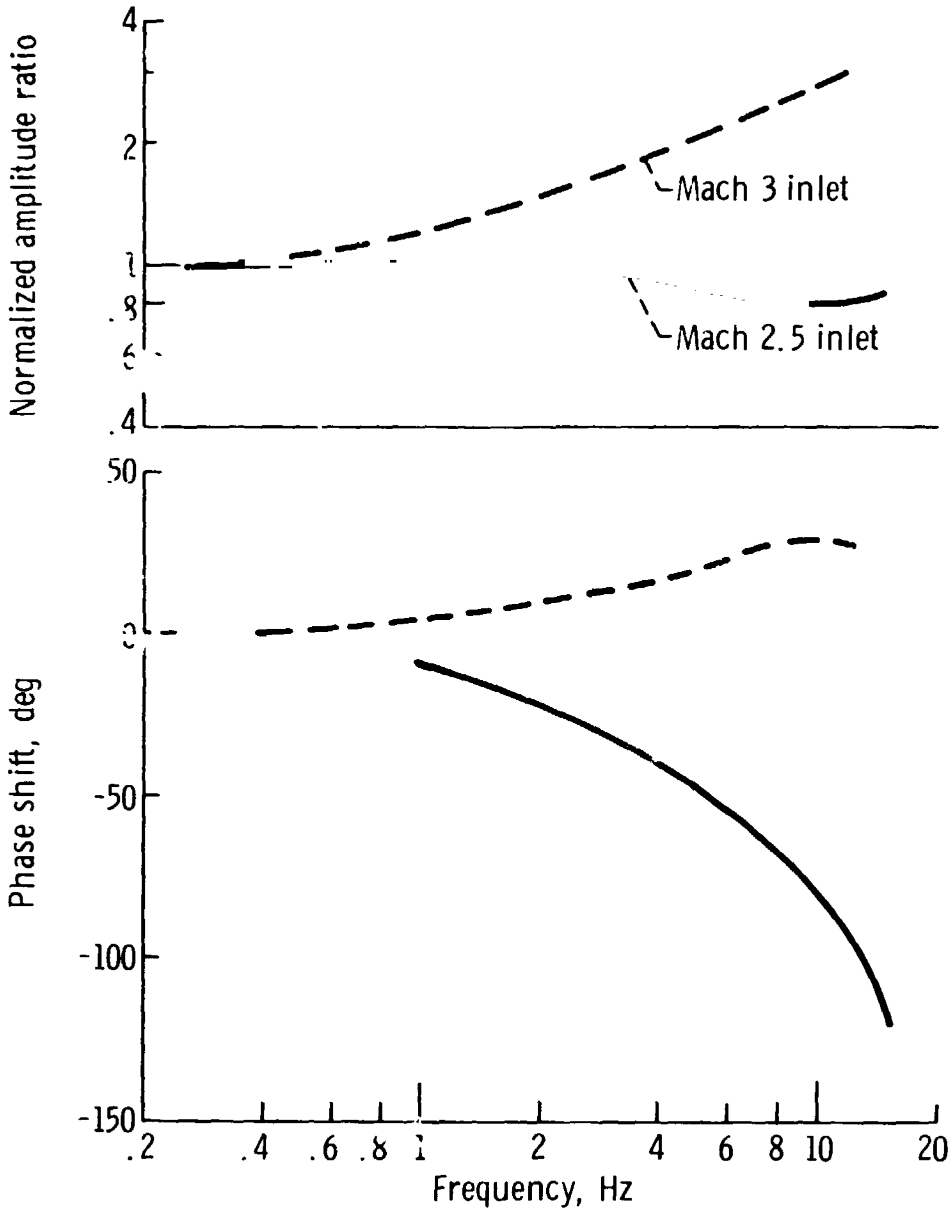


Figure 8. - Shock position response to external disturbance.

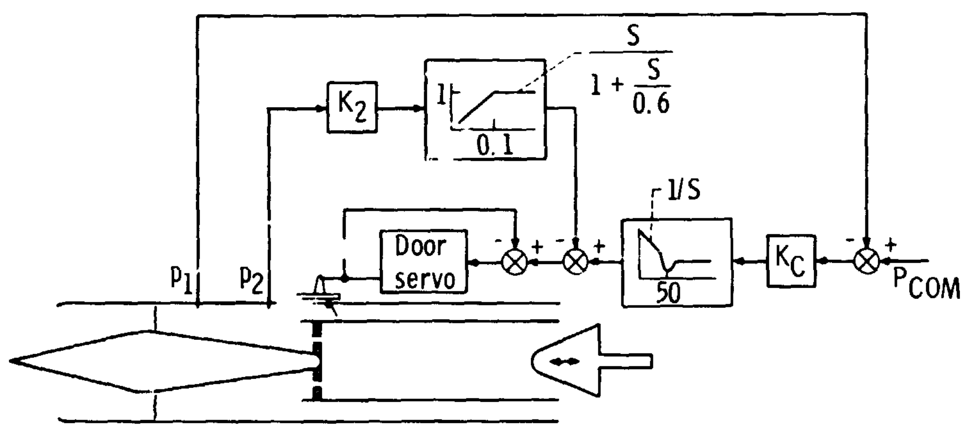


Figure 9. - Inlet bypass door control.

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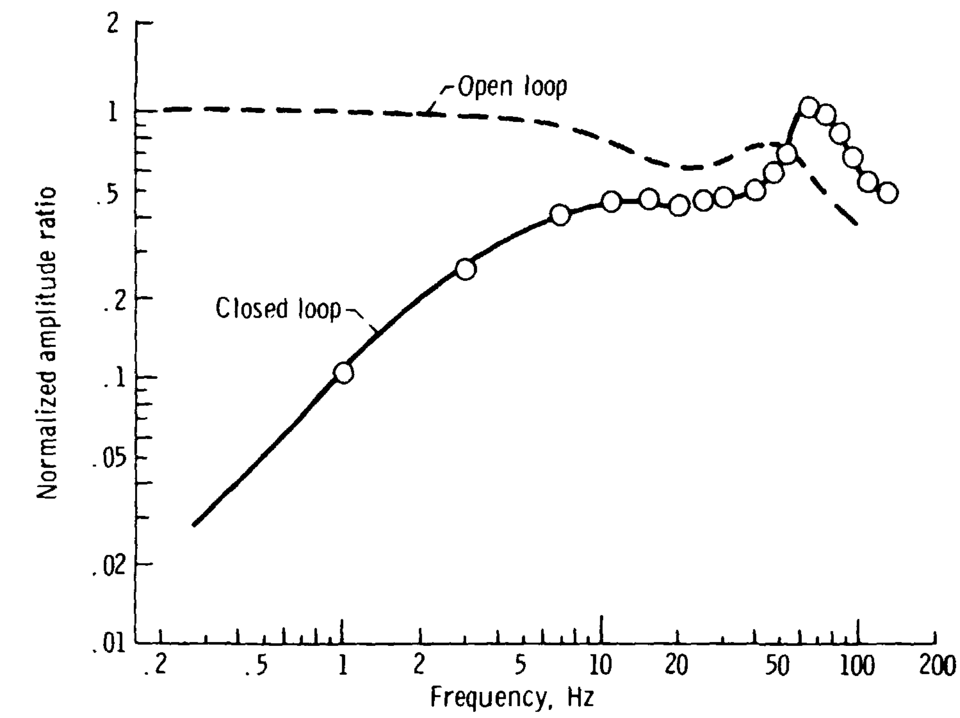

Figure 10. - Control effect on response of terminal shock to downstream disturbance; inlet-engine configuration.

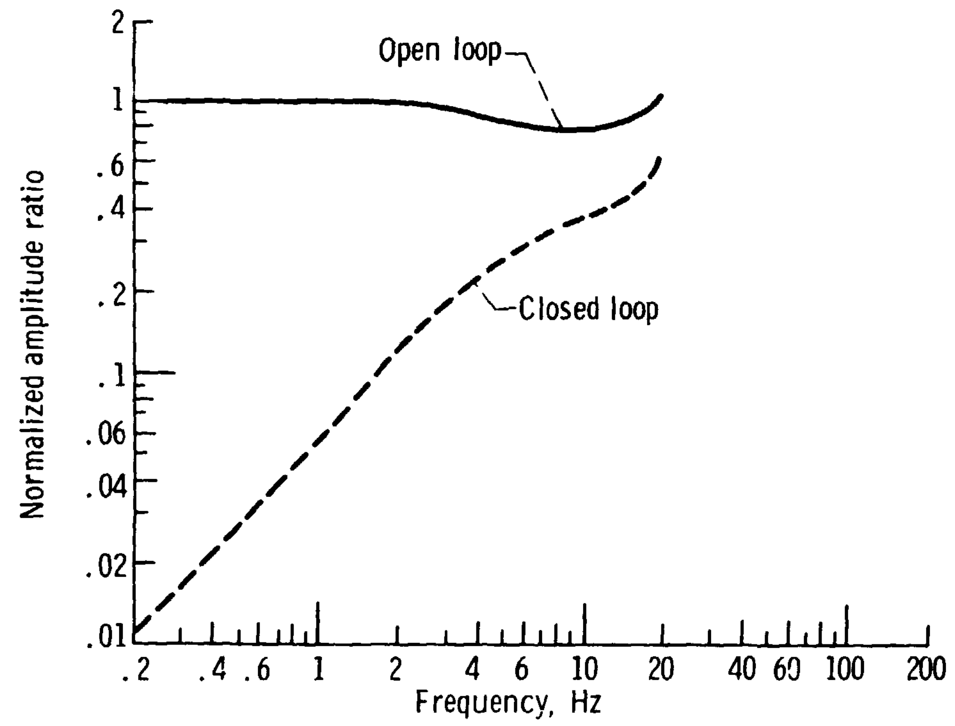

Figure 11. - Control effect on response of terminal shock to upstream disturbance.

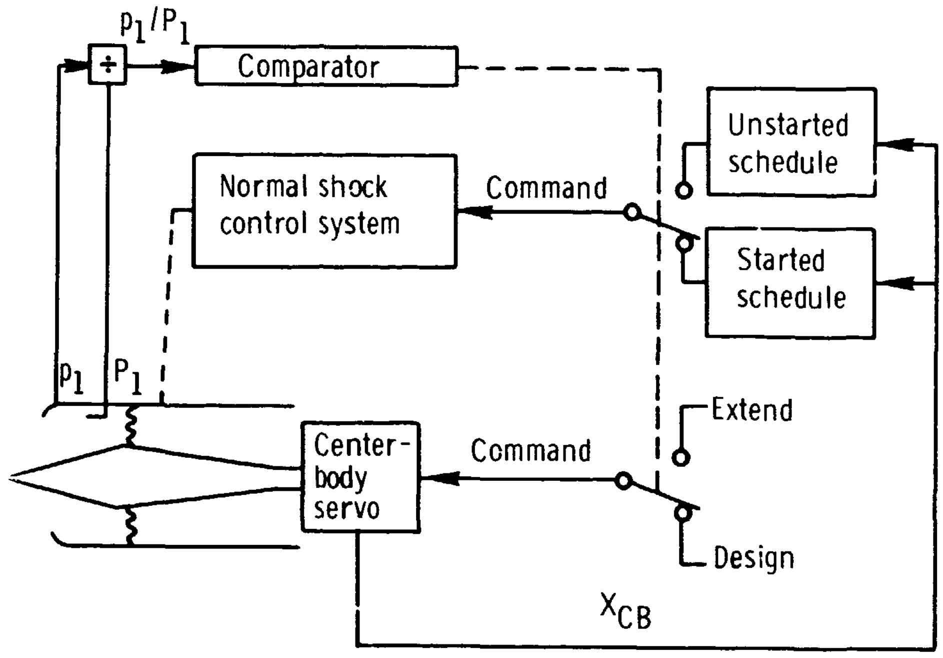

Figure 12. - Restart control.

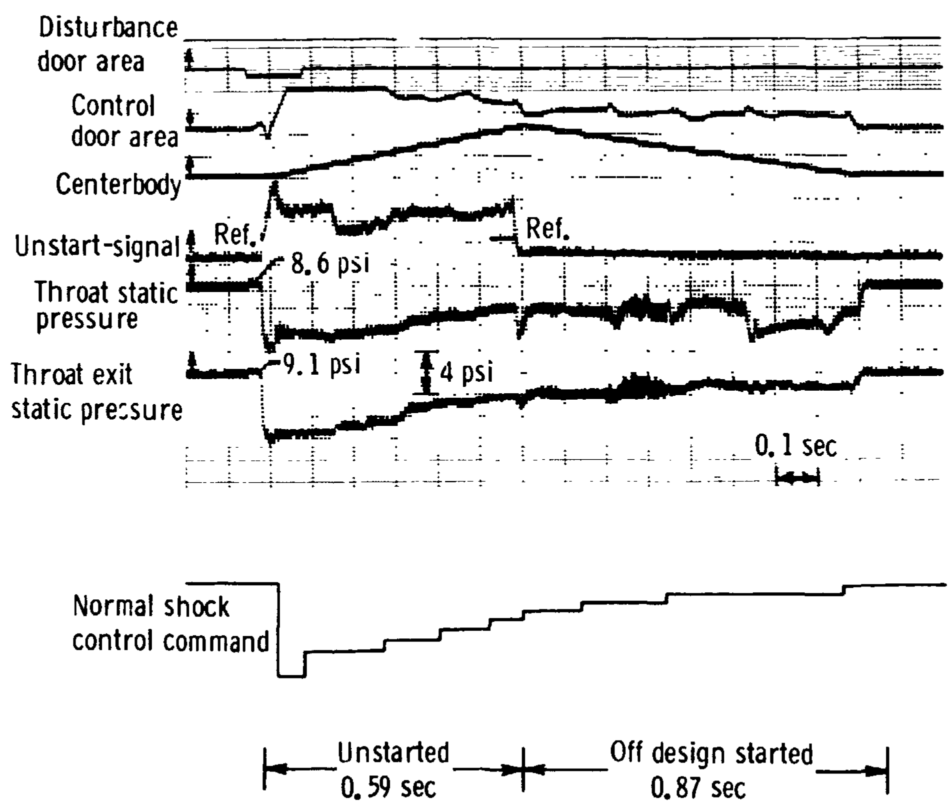

Figure 13. - Unstart-restart transients.

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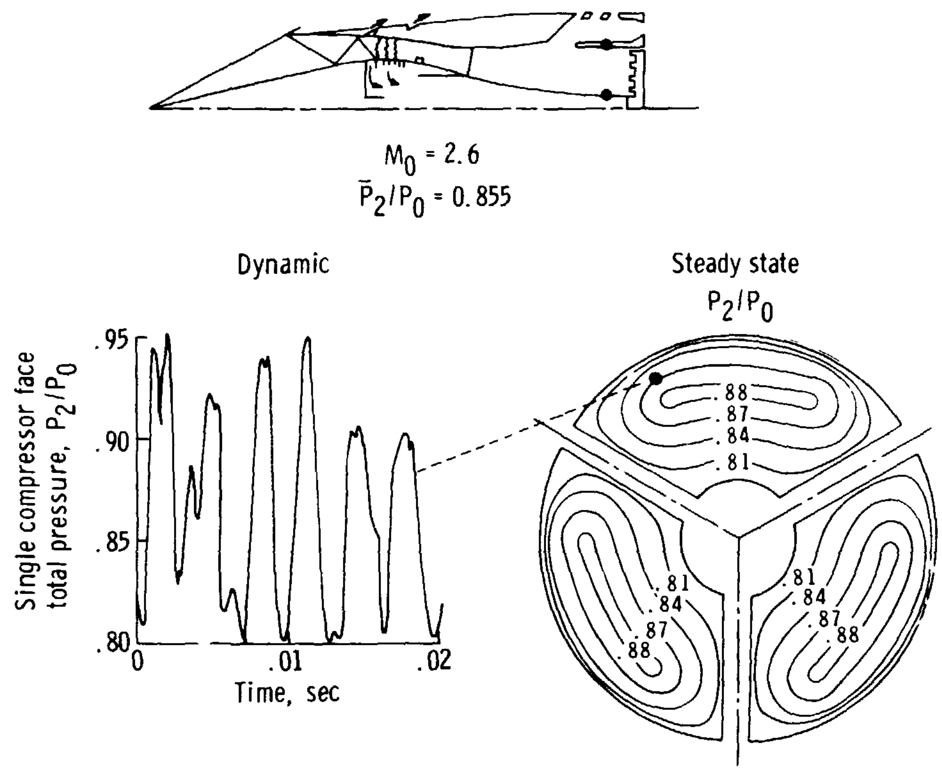

Figure 14. - Inlet distortion.

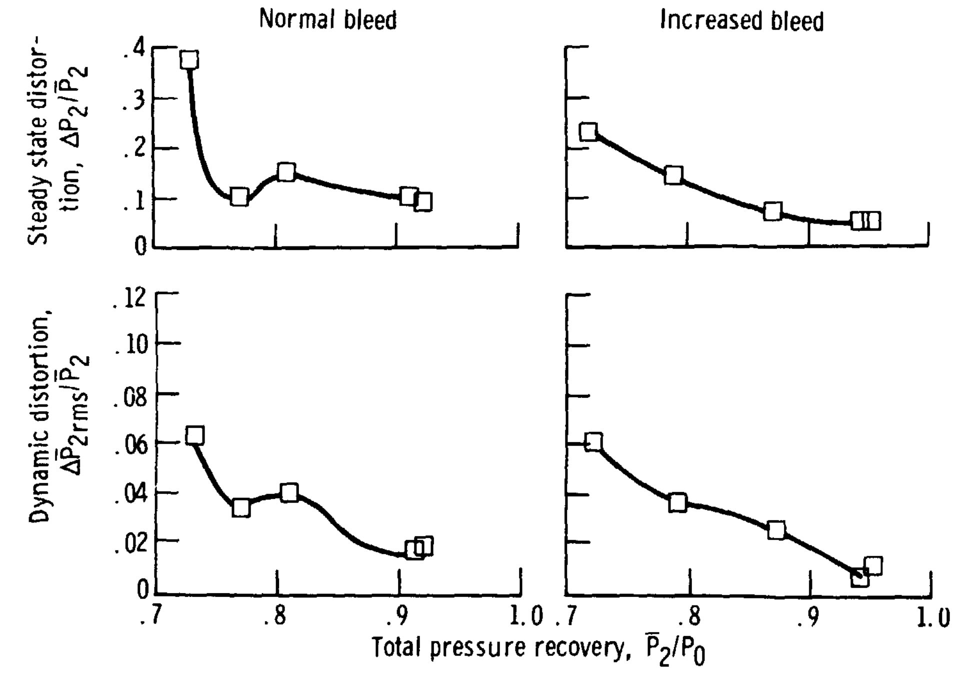

Figure 15. - Effect of increased bleed on distortion; $M_0 = 2.5$; increasing bypass flow.

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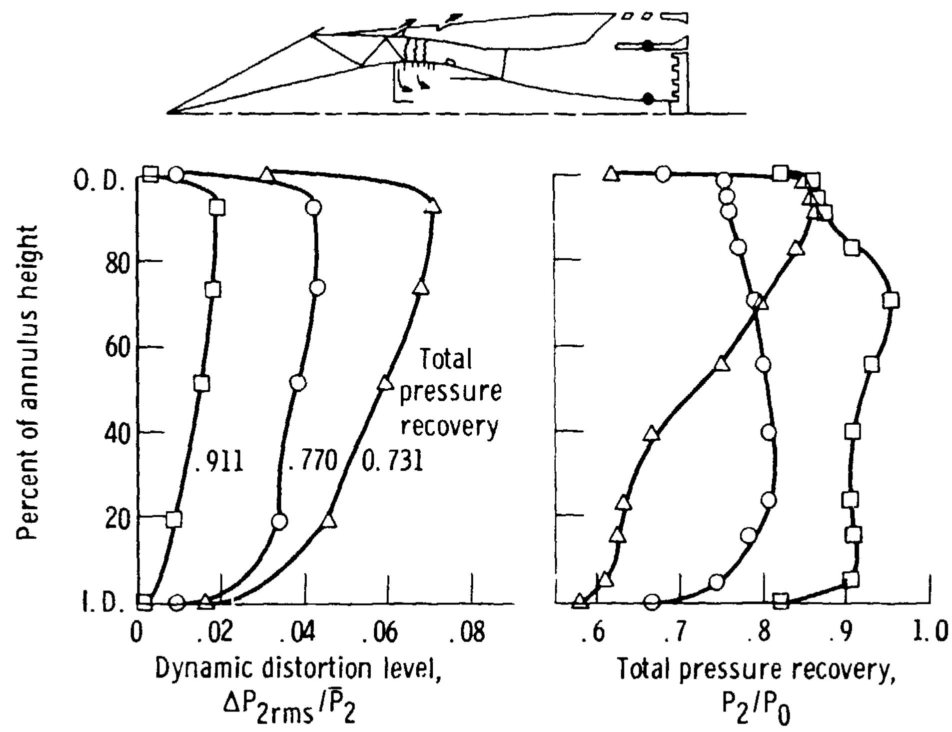
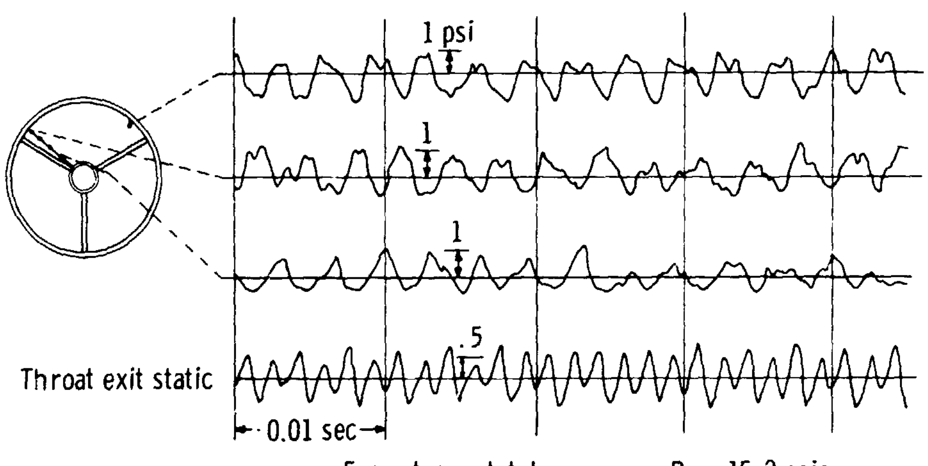


Figure 16. - Radial distortion variation with bypass area.



Free stream total pressure, $P_0 = 15.2$ psia

Recovery, $\bar{P}_2/P_0 = 0.86$

Steady state distortion, $\Delta P_2/\bar{P}_2 = 0.091$

Dynamic distortion, $\Delta\bar{P}_{2\ rms}/\bar{P}_2 = 0.041$

Figure 17. - Typical inlet pressure fluctuations.

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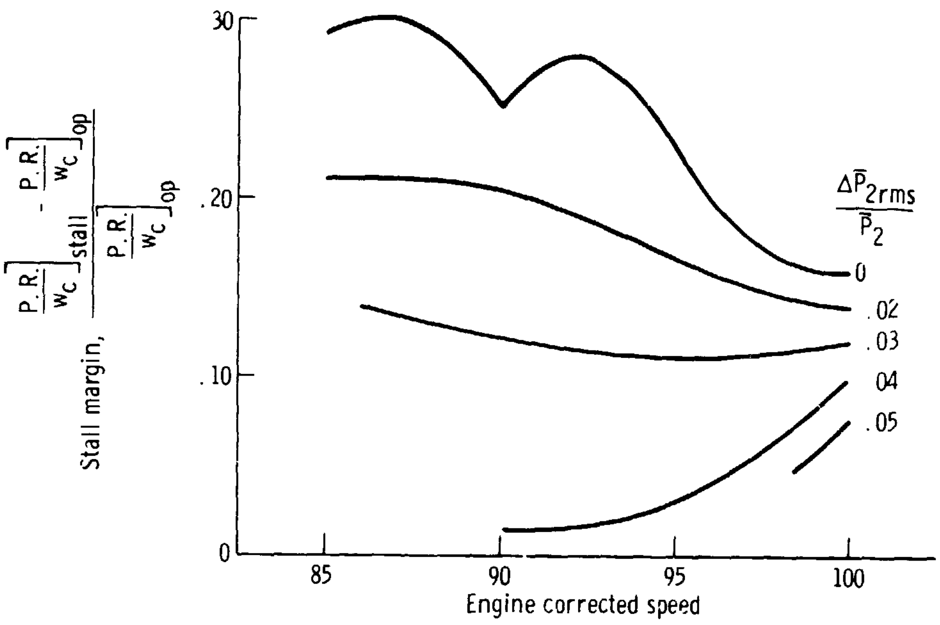

Figure 18. - Distortion effects on compressor stall margin.

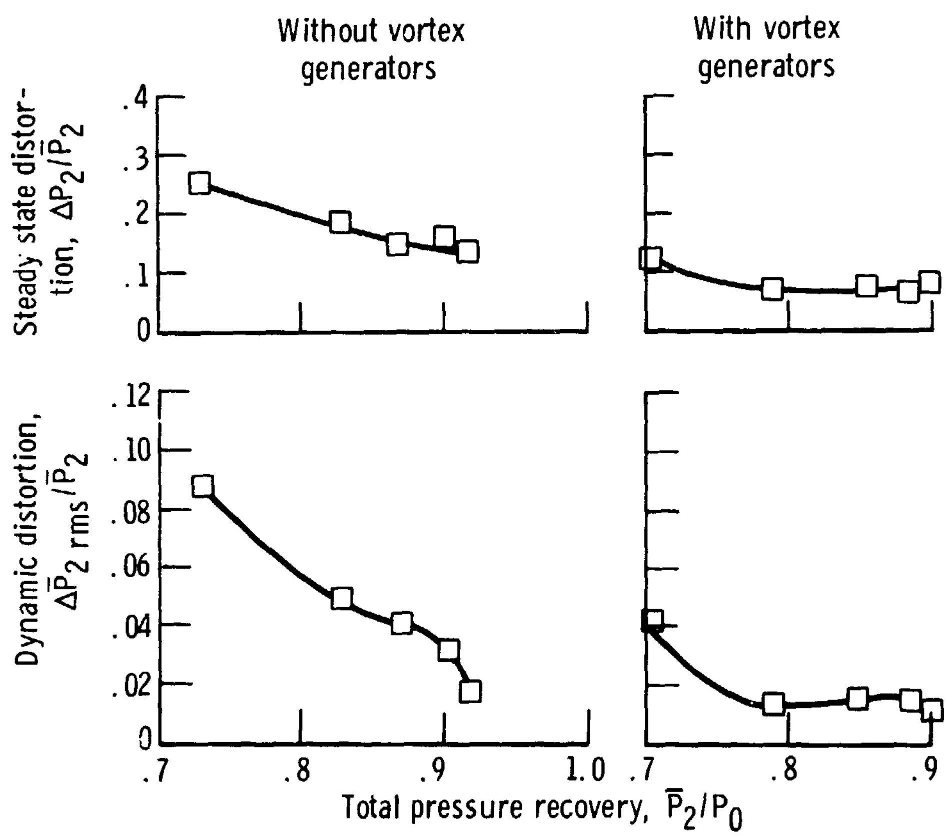

Figure 19. - Effect of vortex generators on dynamic distortion; M_0 = 2.5; increasing bypass flow.

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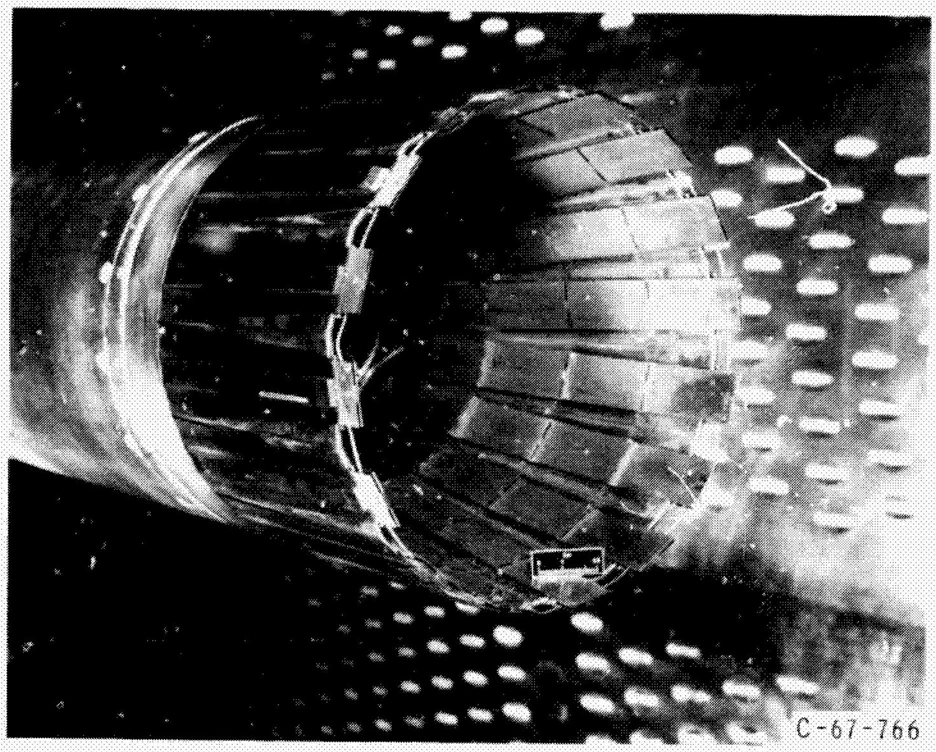
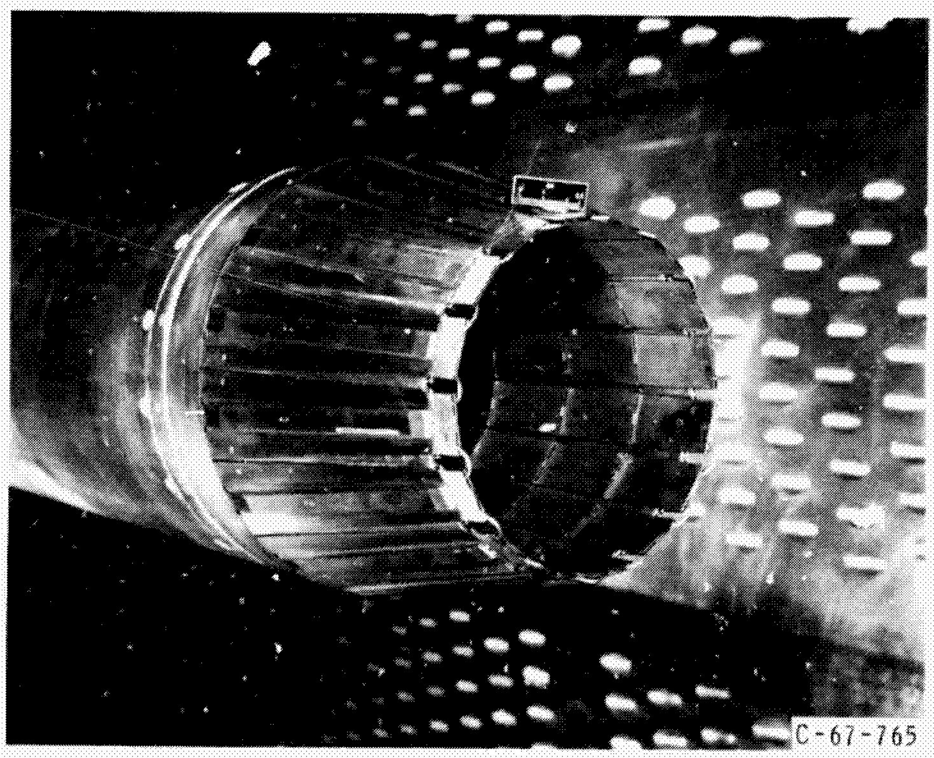


OPEN POSITION



CLOSED POSITION

FIGURE 20. - VARIABLE FLAP EJECTOR.

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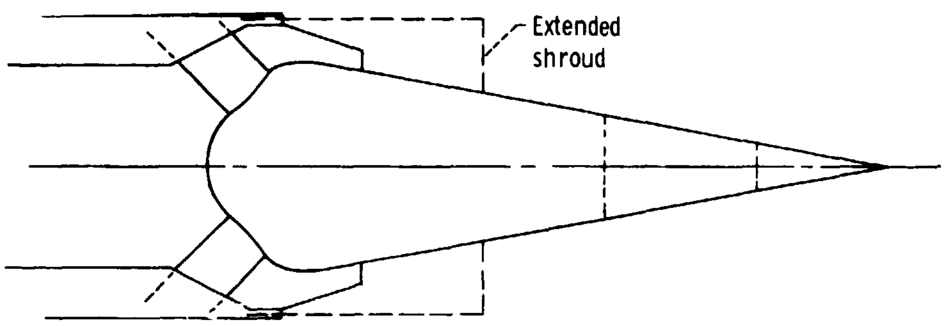

Figure 21. - Conical plug nozzle.

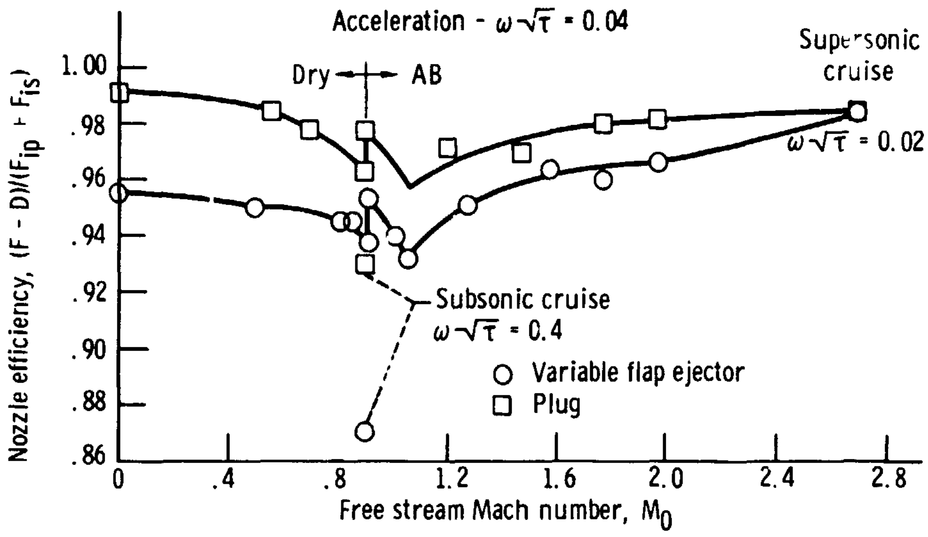

Figure 22. - Nozzle performance; turbojet engine application.

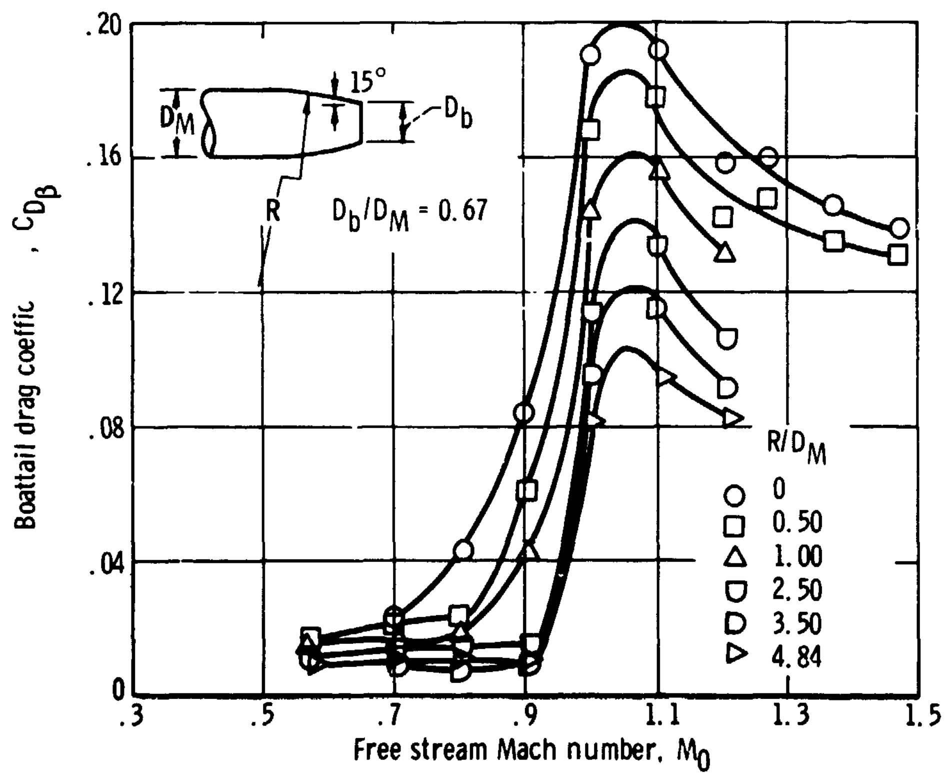

Figure 23. - Effect of afterbody shape on transonic drag rise.

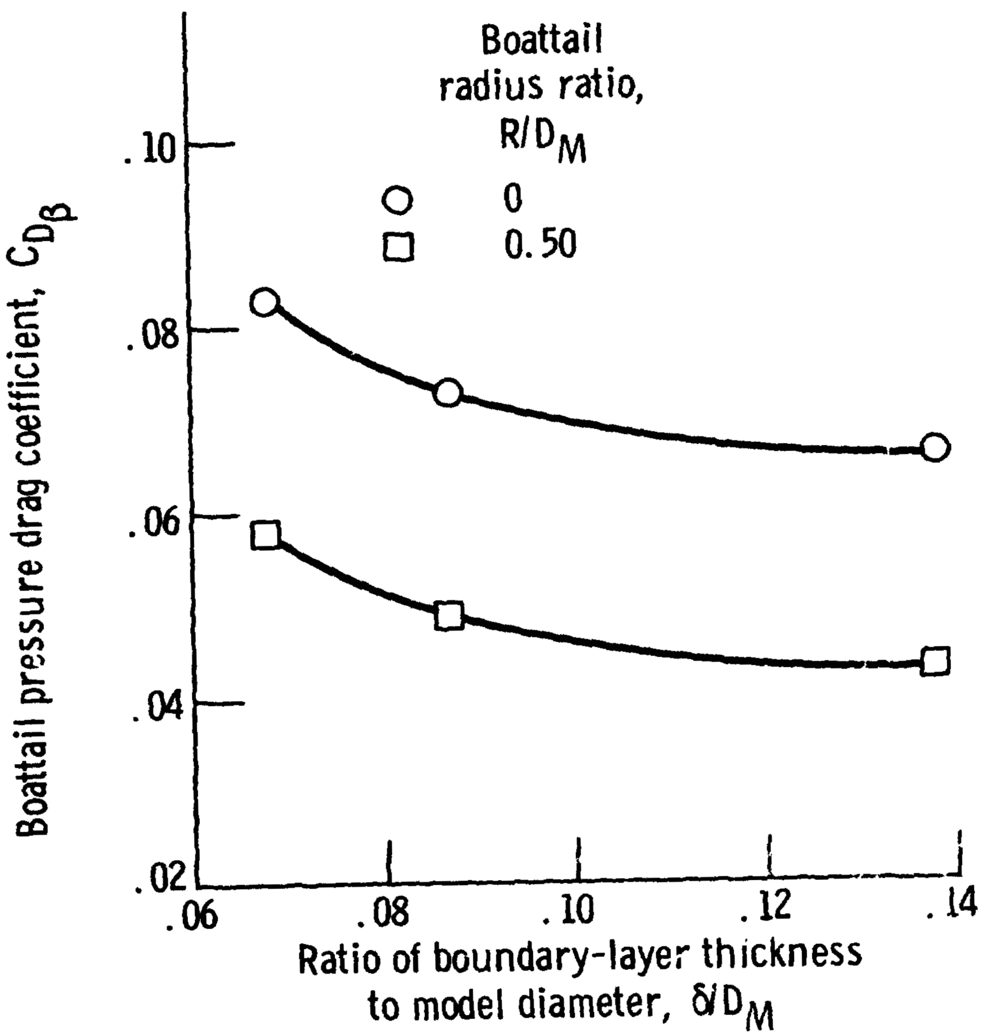

Figure 24. - Effect of boundary-layer thickness on boattail pressure drag; 15° - conical boattails at Mach 0.90.

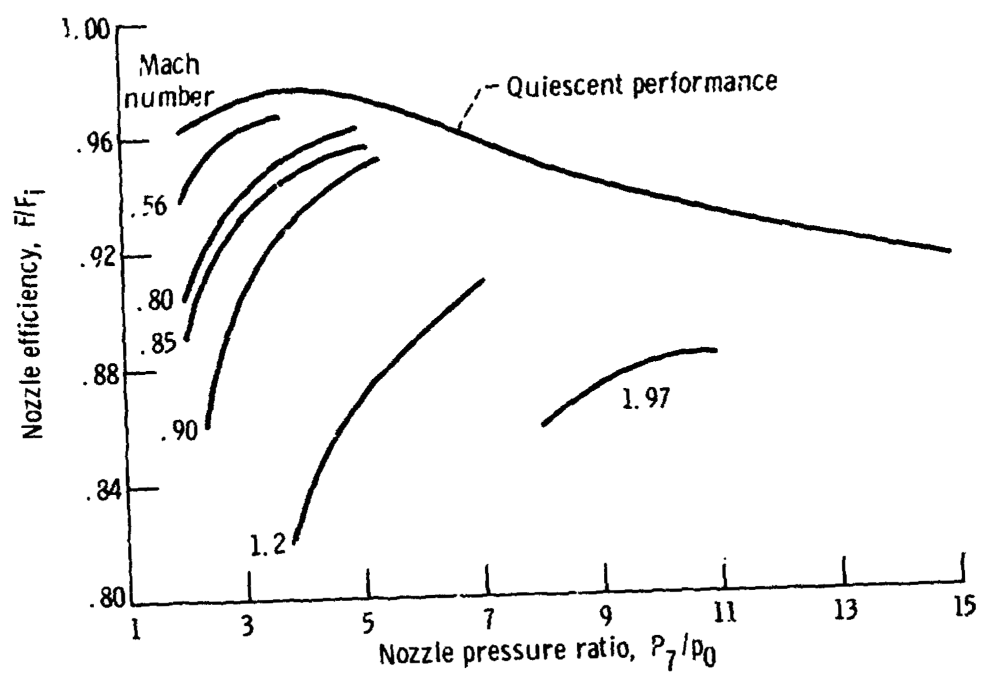

Figure 25. - External flow effect on plug nozzle performance; no reheat; retracted shroud.

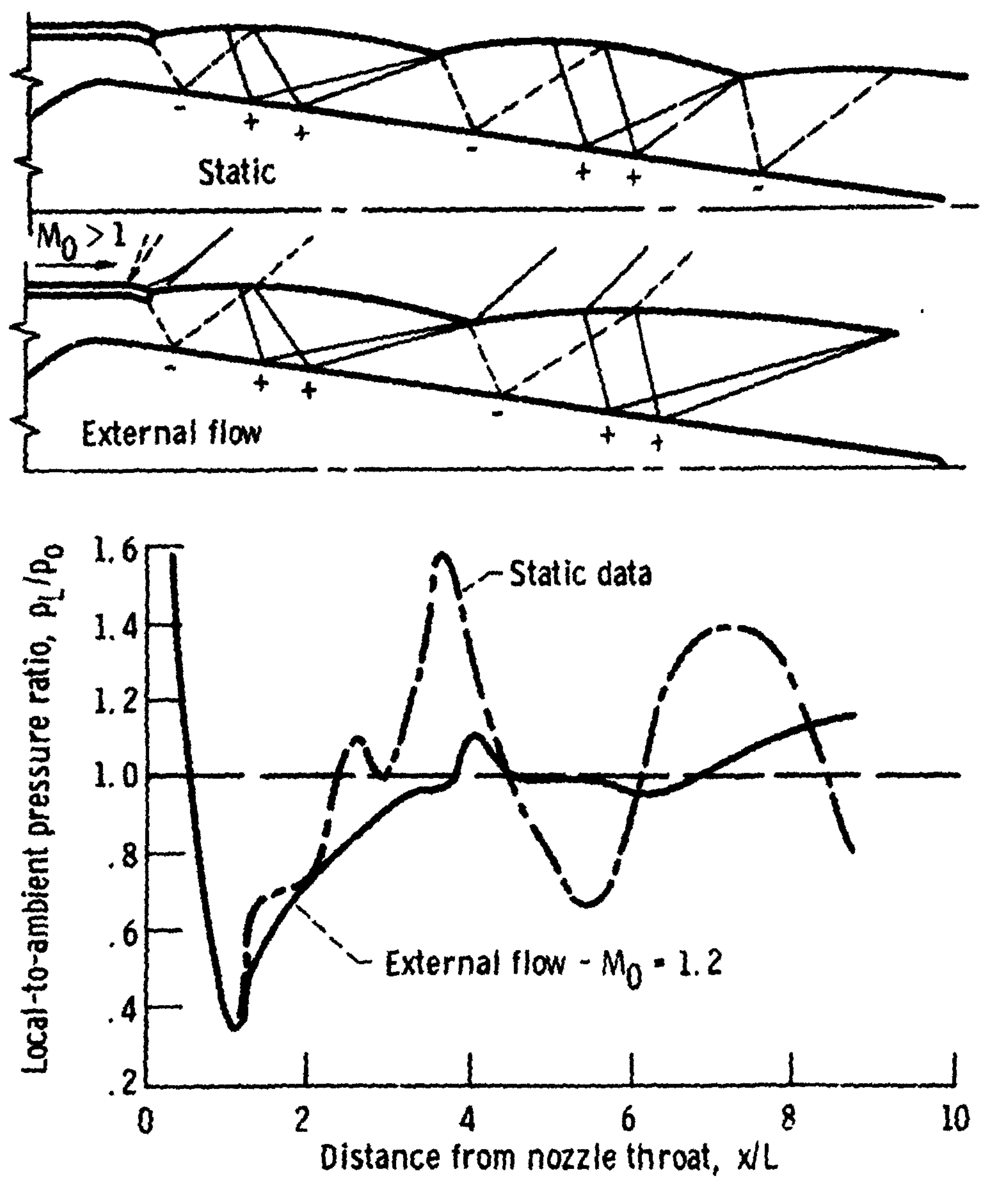

Figure 26. - Effect of external flow on plug pressure distributions; no reheat; retracted shroud, $P_7/p_0 = 5.0$.

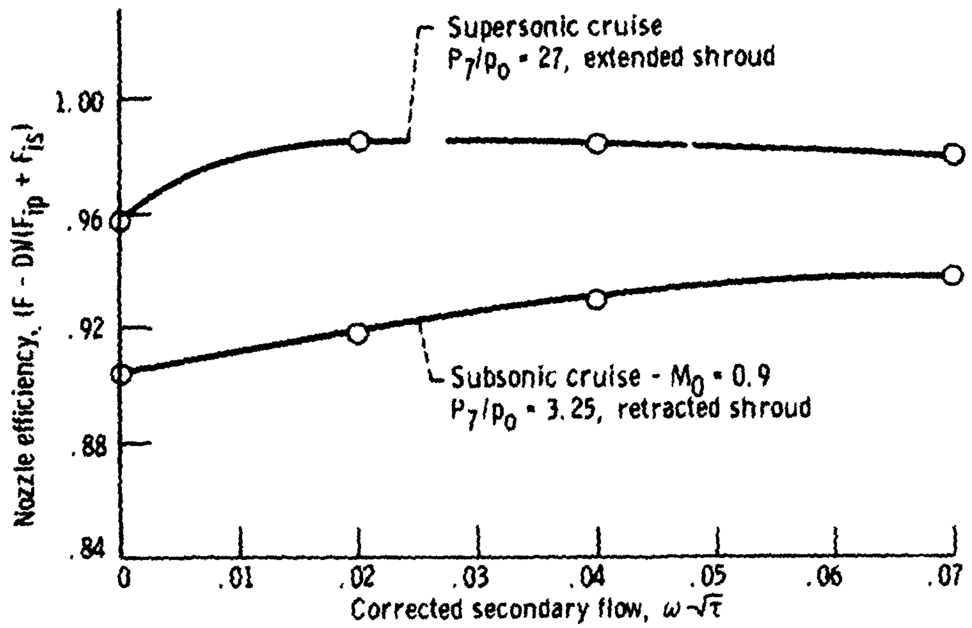

Figure 27. - Effect of shroud secondary flow on plug nozzle performance.

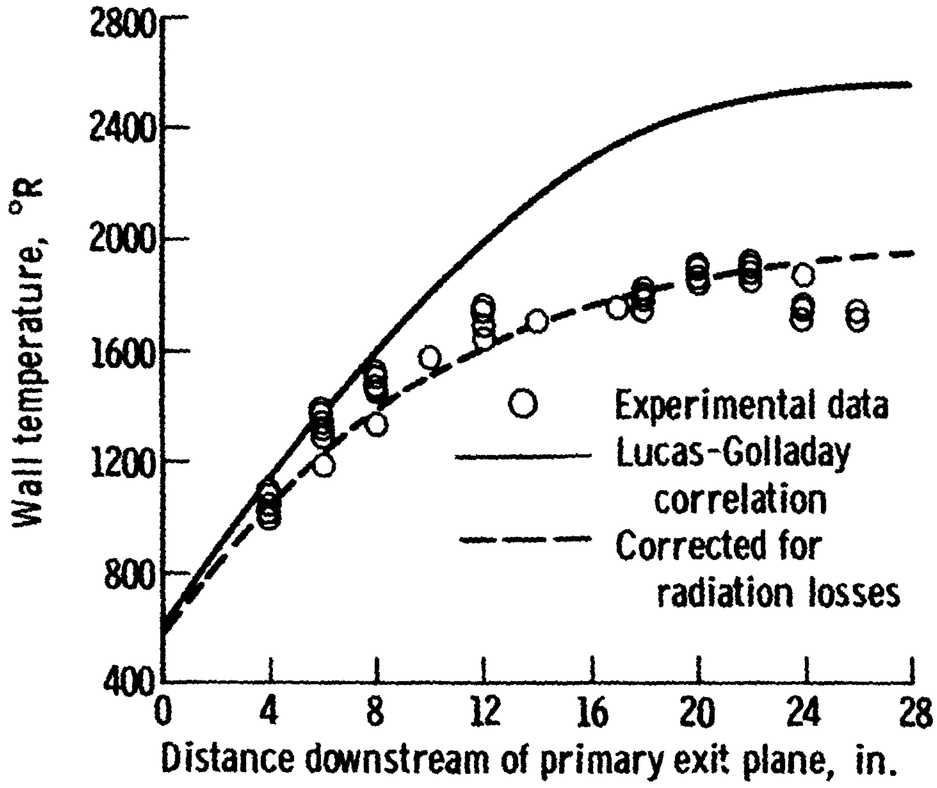

Figure 28. - Temperature distribution along a cylindrical ejector; maximum reheat, $\omega\sqrt{\tau} = 0.05$.

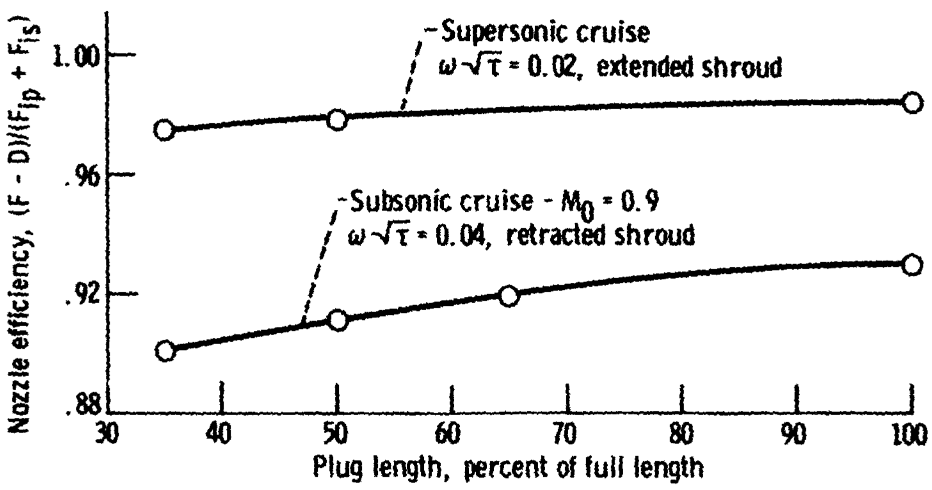

Figure 29. - Effect of plug truncation on nozzle performance.

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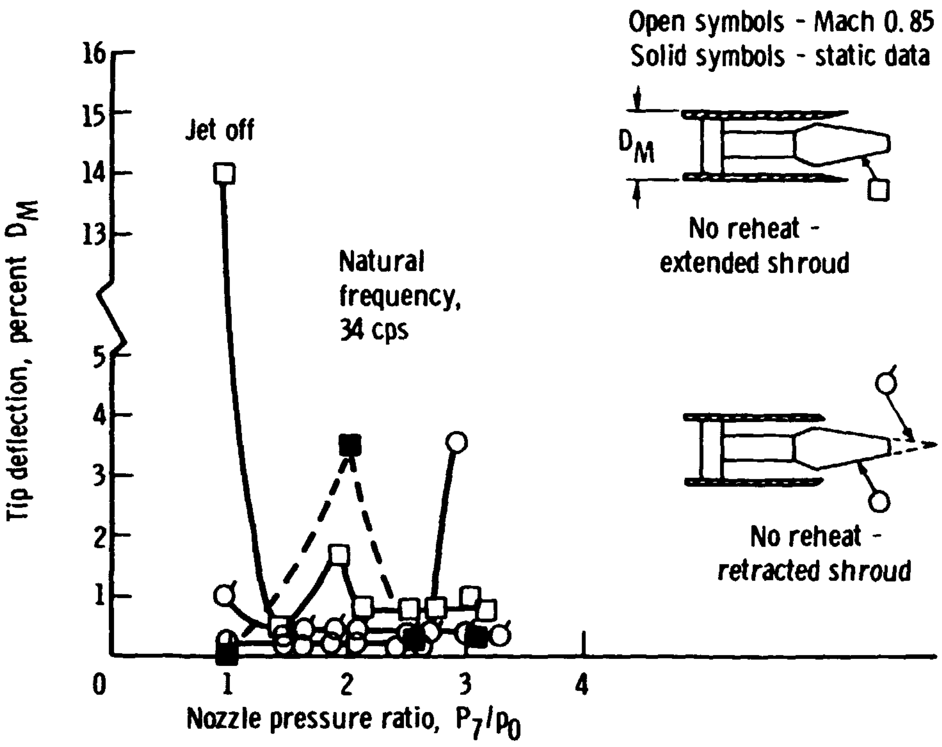

Figure 30. - Observed instabilities with cantilevered plug nozzle.

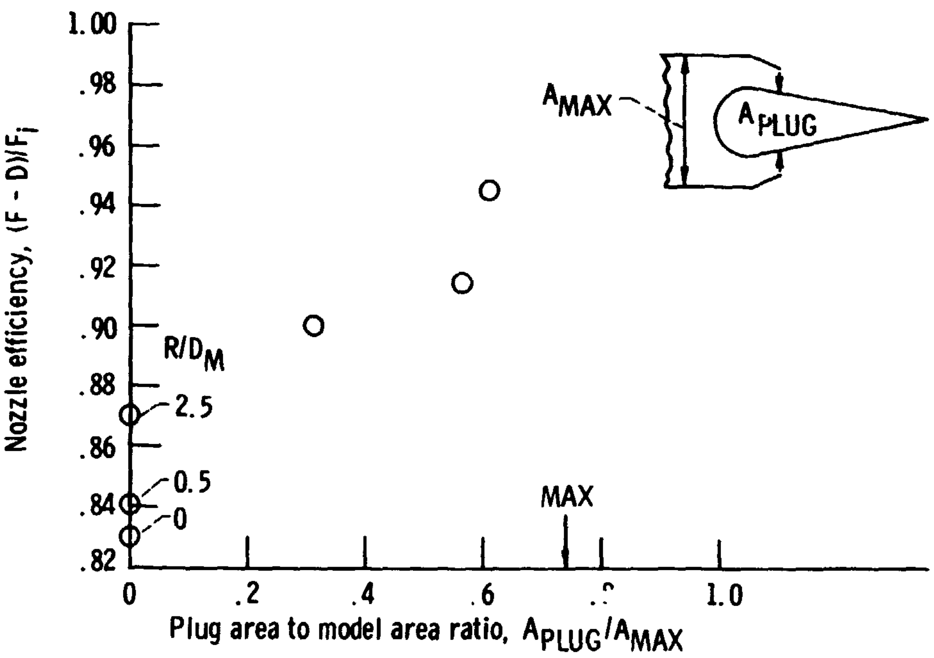

Figure 31. - Effect of plug size on nozzle performance; $M_0 = 0.90$, $P_7/p_0 = 3.25$, $A^*/A_{MAX} = 0.26$.

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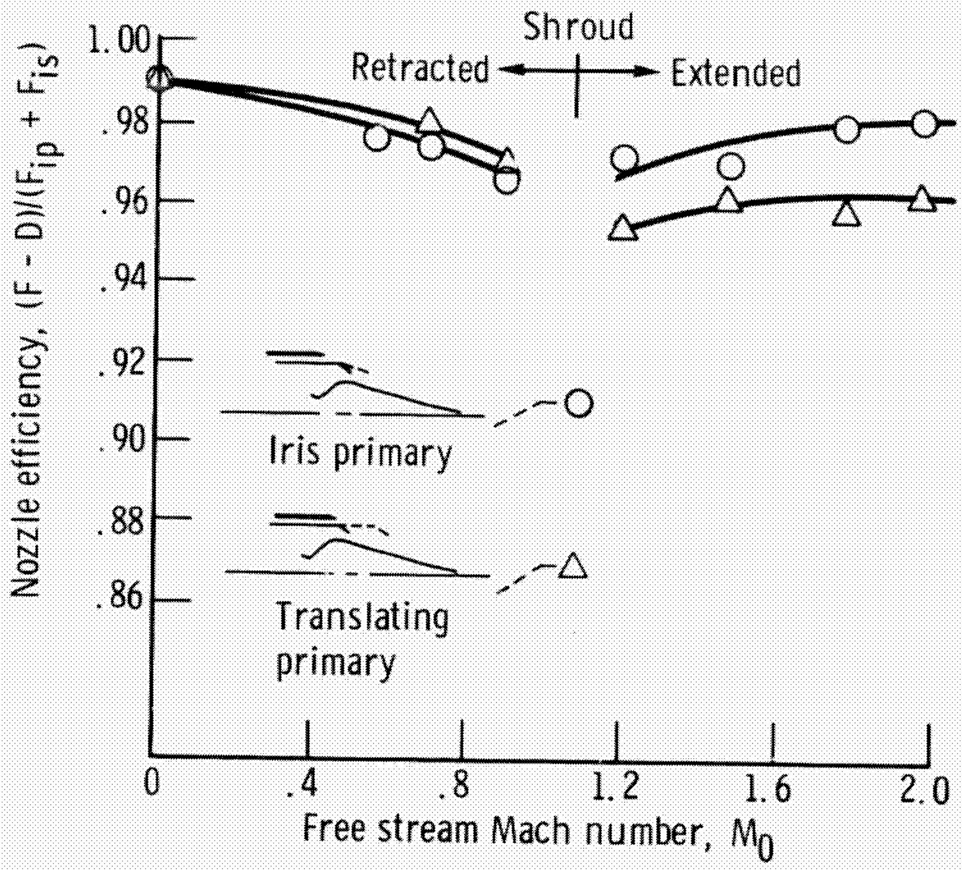

Figure 32. - Comparison of two methods of primary area variation.

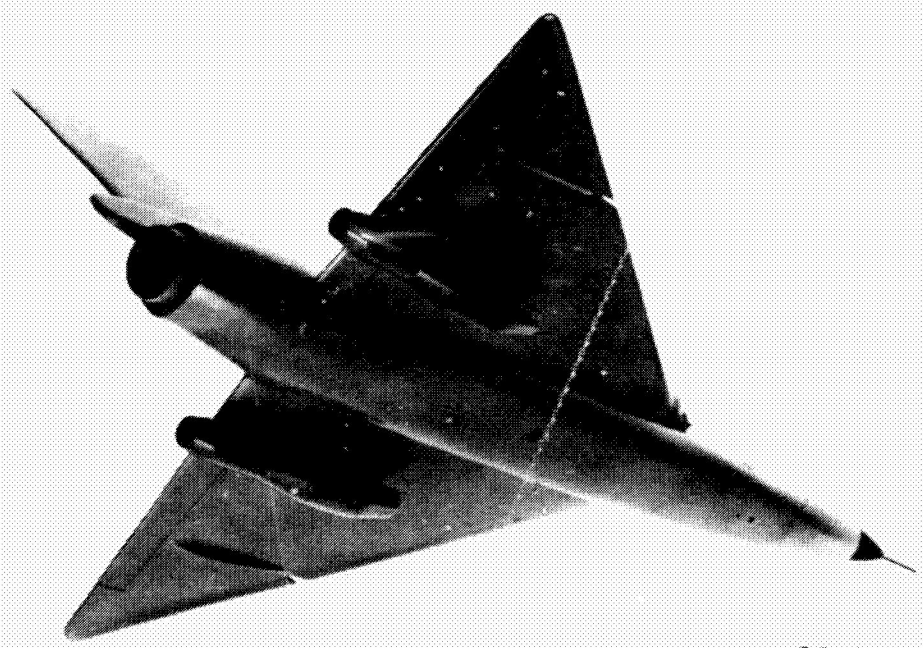

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FIGURE 33. - F-106 NACELLE INSTALLATION.

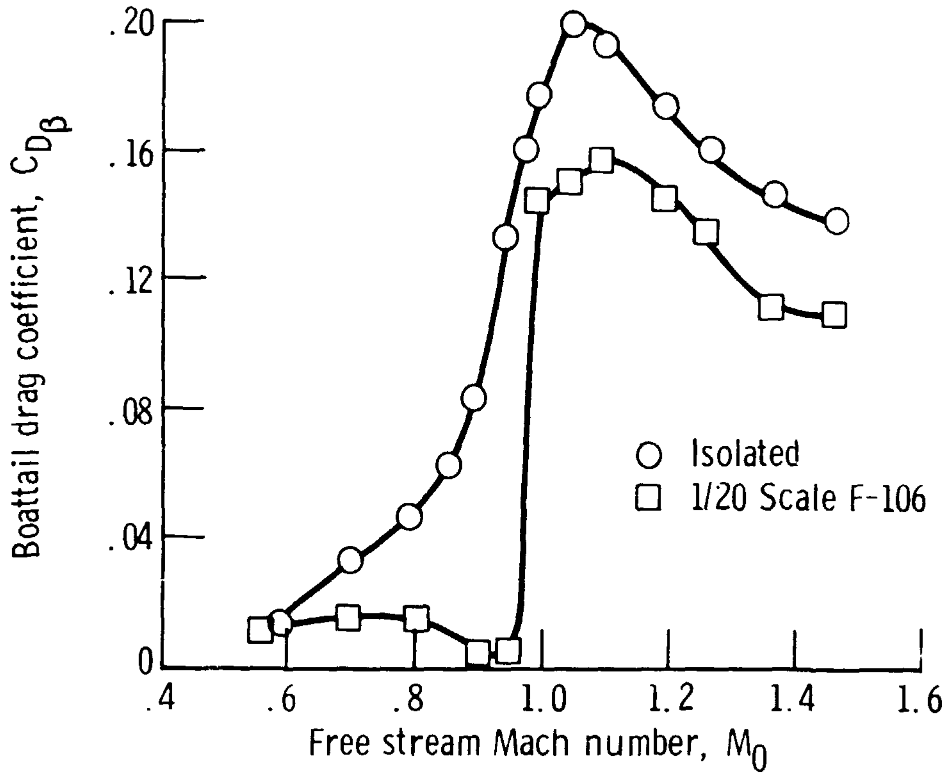

Figure 34. - Installation effect on boattail drag; 15° conical boattail.

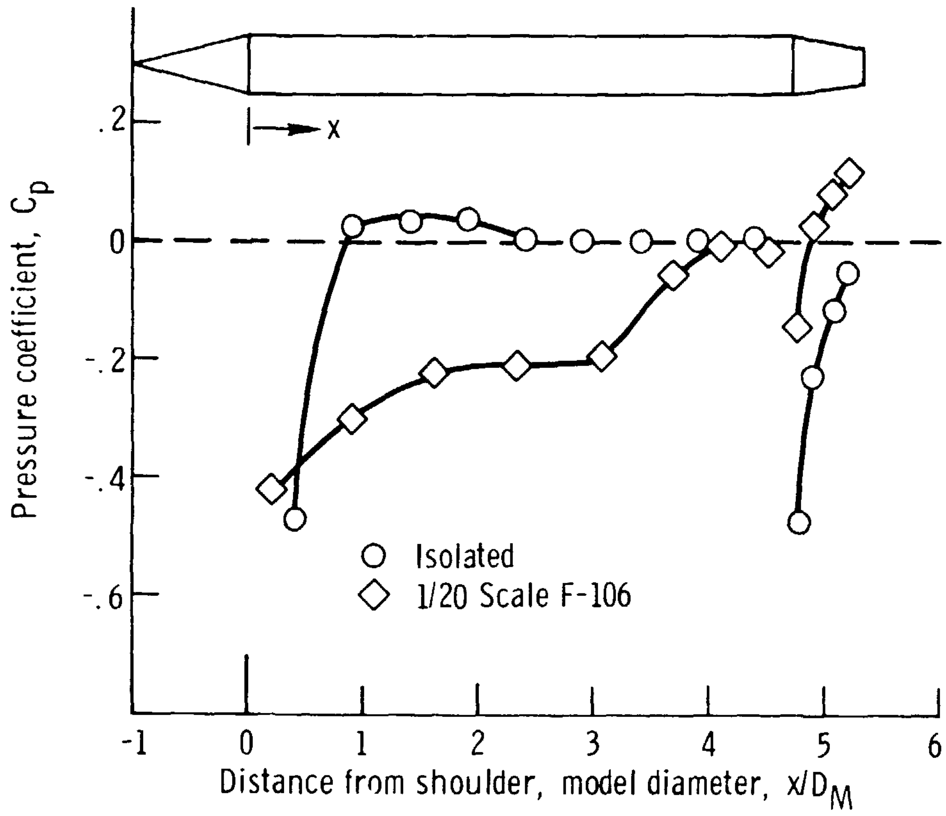

Figure 35. - Installation effect on Nacelle pressures; M_0 = 0.95; 15° conical boattail.

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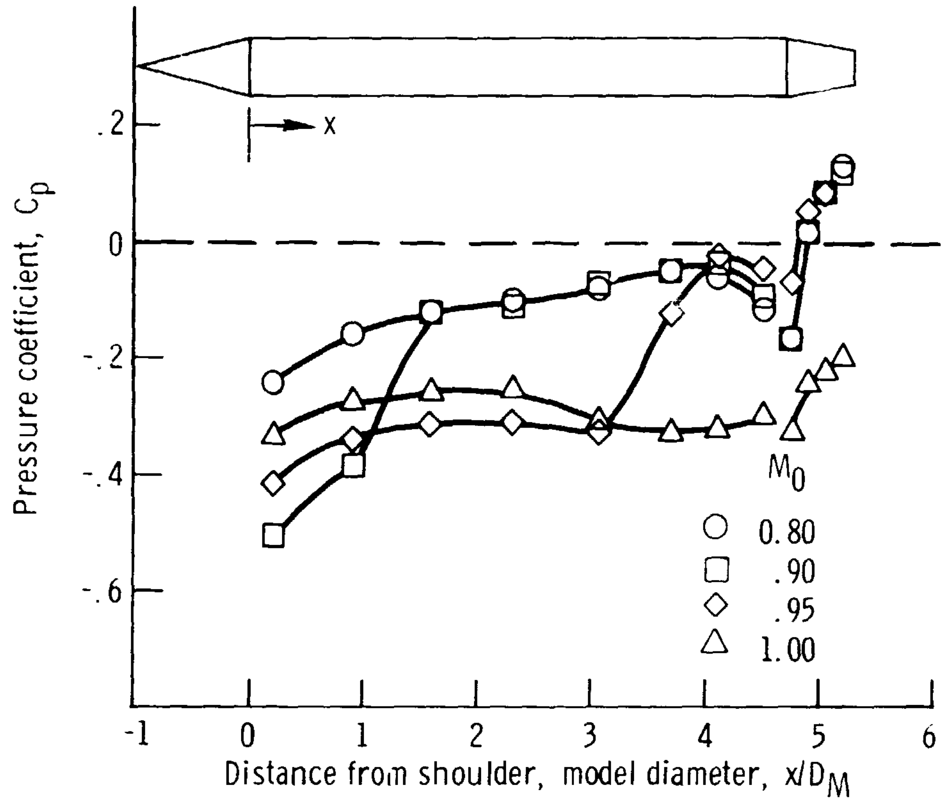

Figure 36. - Effect of Mach number on Nacelle pressures; 1/20 scale F-106 model, 15° conical boattail