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RESEARCH MEMORANDUM

SOME ASPECTS OF SUPERSONIC INLET STABILITY

By James F. Connors

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

SOME ASPECTS OF SUPERSONIC INLET STABILITY

By James F. Connors

SUMMARY

Supersonic inlet stabilization can be generally achieved for Mach numbers up to approximately 2.0 by careful consideration of the possible buzz-triggering conditions. Boundary-layer control and constant-area sections can be effectively utilized on inlets designed to provide stable flow regulation over the entire engine operating range. For Mach numbers above 2.0, the attainment of stability becomes increasingly more difficult as local Mach numbers (and thus normal-shock strengths) increase to aggravate further the shock-boundary-layer interaction problems.

A different approach to the problem of stable flow regulation can be made by assuming that inlets will be generally stable only for limited ranges before becoming inherently unstable. In these cases, variable-geometry techniques seem to provide an adequate solution.

It has also been demonstrated that the engine itself can, in some instances, exert a stabilizing influence on the inlet. Further definition of this effect is needed with full-scale inlets and more advanced engines.

INTRODUCTION

At supersonic speeds, the inlet-buzz condition is characterized by large pressure and mass-flow oscillations which must be avoided or attenuated for satisfactory engine operation. Otherwise, the attendant flow pulsations could result in flameout in the combustor or even structural damage to the engine. In most instances, the origin of inlet buzz can be traced back to either of two triggering mechanisms; (1) the vortex sheet or slipline intercepting the cowl lip or (2) compression-surface flow separation. Both of these are quite similar in principle and have been recognized for some time. In each case, the initiation of buzz is distinguished by a sudden change or discontinuity in the total-pressure profile at the diffuser entrance with a subsequent tendency towards separation of the internal flow.

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SYMBOLS

The following symbols are used in this report:

h	height of boundary-layer diverter
M	Mach number
M_0	free-stream Mach number
m_c	critical mass flow entering inlet
m_i	injection mass flow
m_o	maximum mass flow that can enter inlet
α	angle of attack
δ	boundary-layer thickness
θ_l	cowl-position parameter
θ_s	conical shock angle

4006

DISCUSSION

Inlet Flow-Stabilization Techniques

In the past, sporadic success in attenuating buzz has been achieved through the use of constant-effective-area or zero-diffusion throat sections (ref. 1). This scheme allows the entrance flow with its discontinuous profile to mix before undergoing subsonic diffusion. With nose inlets, buzz can also be generally avoided for Mach numbers up to 2.0, first, by observing the slipline criterion of reference 2 (usually by positioning the oblique shock slightly inside or well ahead of the cowl) and, secondly, by using compression-surface angles which are large enough to keep the local Mach number below the normal-shock value of approximately 1.3 which is required for separation of a turbulent boundary layer (ref. 3). Near Mach 2.0, however, the design of high-compression multiple-shock inlets dictates the use of initially smaller compression-surface angles and correspondingly higher surface Mach numbers. In these cases, boundary-layer-control techniques, such as illustrated in figure 1, can be utilized.

The data in figure 1 were obtained at a free-stream Mach number of 1.9 with the double-cone axisymmetric nose inlet of reference 4. This configuration employed two conical compression surfaces with half-angles of 20° and 28° and with corresponding supercritical surface Mach numbers of 1.49 and 1.2, respectively. Supercritically, the oblique shocks were located just inside the cowl. As the normal shock moved subcritically upstream of the cowl, the slipline (indicated by the dashed line) did not move across the cowl lip, and there were no adverse effects due to interaction of the bow shock with the second-cone boundary layer. With no boundary-layer control, stable subcritical operation was obtained for a range of mass-flow ratios down to approximately 0.7. At this minimum stable condition, the normal shock was located at the break between the two conical surfaces. Simultaneously, with the onset of buzz and as the bow shock moved out on the first cone, the boundary layer was observed to lift off the surface and separate. This separation was, of course, due to the increased surface Mach number on the first cone.

In this case, where to apply boundary-layer control was clearly defined. The centerbody was vented to ambient pressure and two double rows of holes were installed on the first cone. With boundary-layer suction thus applied, stable subcritical operation was obtained for a range of mass-flow ratios down to approximately 0.1. At the minimum stable condition, the bow shock stood upstream of the bleed holes. With suction, however, the critical pressure recovery was reduced from 0.92 to about 0.9; apparently, the bleed holes created additional supersonic losses. In both cases, the supercritical mass-flow ratio was unity. The maximum bleed flow was estimated at approximately 1.5 percent of critical mass flow.

The effect of angle of attack on both pressure recovery and mass flow is illustrated by the data of figure 2. Angle of attack generally caused reduction in both pressure recovery and stable mass-flow range. The stable operating range of the inlet is indicated by the cross-hatched areas for the no-suction and suction cases. As the inlet goes to angle of attack, the compression-surface Mach numbers decrease on the windward side and increase on the leeward side. At the higher angles, the second-cone Mach number was thus sufficiently increased on the leeward side that the interaction between the bow shock and the boundary layer was no longer satisfactory and the accompanying separation was enough to trigger buzz prematurely. In this particular case, stability might have been improved still further at the higher angles of attack if additional suction had been applied on the leeward side of the second cone. Thus, in this Mach number range, boundary-layer suction can, in some cases, be effectively utilized to obtain stable flow regulation.

For free-stream Mach numbers considerably above 2.0, the local-surface Mach numbers correspondingly increase, and the most effective location of boundary-layer control from a stability viewpoint is no longer clearly defined. At these high Mach numbers, the point of incipient separation will, of course, vary with diffuser-normal-shock position since the upstream surface Mach numbers are now everywhere greater than a critical normal-shock separation value (which is again approximately 1.3). Very little data are available on inlet stability near Mach 3.0. In one case, at least, some degree of success has been achieved with mass-flow injection (or boundary-layer energizing) on a half 2-cone side inlet at a free-stream Mach number of 2.96 (see ref. 5).

The results of this study are summarized in figure 3. The inlet utilized two semi-cones with angles of 20° and 34° with corresponding supercritical surface Mach numbers of 2.25 and 1.71, respectively. A gap was provided between the first and second cones for injection of high pressure air parallel to the second compression surface. In a flight application, this injection air could be supplied, for example, by compressor bleed. Sketches of typical minimum-stable-mass-flow patterns with and without injection are shown on the figure. Performance results are summarized in the table. With the inlet out of the boundary layer ($h/\delta > 1.0$) there was no stable subcritical range without flow injection. However, with an injection mass-flow ratio of 0.02, the subcritical stability range was equal to 24 percent of critical mass flow. A total-pressure-recovery decrement of 0.04 was encountered due only to the change in geometry (that is, the provision of the injection gap). Actually, the critical pressure recovery falls off quite markedly as the inlet is submerged in the boundary layer. For an $h/\delta \approx 0.26$, an injection mass-flow ratio of 0.04 increased the subcritical stability range from approximately 7 to 49 percent of critical mass flow.

4006

Variable-Geometry Techniques for Stable Flow Regulation

The techniques discussed so far have been directed towards the development of inlets that would provide stable flow regulation over the entire operating range. Actually, in a typical supersonic flight application, the turbojet engine can have two distinct operating areas which require stable regulation. The first is for a limited range at high mass-flow ratios and occurs during transient operation, for example, during wind gusts or an overshoot of the controls system. Here, thrust must be maintained. Consequently, stable flow regulation must be accomplished without excessive loss in recovery or increase in drag. The second operating area occurs during throttle-closure to engine-idle airflow setting. For this condition, stability can be attained with little regard for loss in recovery or increase in drag, since the aircraft is to undergo rapid deceleration.

With two such modes of operation, another approach can be made to the problem of attaining stable flow regulation. In this method, inlets are assumed to be generally stable only for a limited mass-flow range before becoming inherently unstable. In this case, variable-geometry techniques as illustrated in figure 4 can provide an adequate solution.

Variable geometry, for example, in the form of a translating spike or a bypass arrangement, can be effectively used at the higher mass-flow ratios. Here, the normal shock is maintained at the throat, while the reduced air-flow requirements of the engine are met by supersonic spillage behind an oblique shock or by spillage through the bypass. Thus, buzz-triggering conditions at the cowl lip may be avoided altogether.

At the lower mass-flow ratios, or for flight conditions corresponding to throttle closure where pressure recovery is not too important, spoiler
4009 techniques may find some application. These techniques largely involve the use of variable-geometry devices to force a bow shock to stand well upstream of the cowl with attendant large mass-flow spillage rates. The actual form of such spoilers can be quite varied. In the axisymmetric case, variable flaps or projections moving out of the compression surfaces might conceivably be employed to detach the flow and force a bow wave ahead of the inlet. Two-dimensionally, such a technique has been effectively demonstrated by means of a variable-second-ramp side inlet for Mach numbers from 1.5 to 2.0 (see ref. 6). The results for $h/\delta > 1.00$ are shown in figure 5. This particular inlet geometry permits an increase in the second-ramp angle to values in excess of the local shock-detachment values. Thus, for stable operation at low mass flows (for example, where engine-idle conditions correspond to mass-flow ratios of approximately 0.4) this scheme proved quite satisfactory. Data are shown for only two second-ramp positions - the 18° ramp representing the design operating position and the 30° ramp representing the detachment or low-mass-flow condition. At each Mach number, stable operation was obtained for mass-flow ratios in the vicinity of 0.4, the hypothetical engine-idle condition. At Mach 2.0, the data for the two second-ramp positions do not overlap with respect to stable mass-flow range; however, it might be anticipated that the intermediate ramp positions would provide a continuous transition of stable operation down to the engine-idle mass flow.

Another technique for attaining stability, but at the expense of recovery, consists of retracting the compression surface and positioning the oblique shock well inside the cowl lip. This method can be demonstrated with a translating-spike inlet configuration. As illustrated in figure 6, a cowl-position parameter θ_l will be used to define the range of spike translation. This parameter is the angle between the inlet axis and a line from the spike tip to the cowl lip. The design position is that point where θ_l equals the conical shock angle. As shown in figure 7 for a single-cone axisymmetric nose inlet at a Mach number of 2.0 (ref. 7), large stable mass-flow ranges were obtained with values of cowl-position parameter of 2° to 3° greater than the design shock-on-lip value. In this case, the position of the oblique shock well inside the cowl prevents the slipline from intercepting the cowl lip. As the tip shock is moved inside (that is, increasing θ_l from the design value), the stable operating range increases quite markedly and, correspondingly, critical pressure recovery decreases. For values 4°

greater than the design value, stable operation can be obtained to the hypothetical engine-idle condition (mass-flow ratio ≈ 0.4). As the tip shock is moved outside (that is, decreasing θ_l from the design value), no significant increase in the stable range occurs; however, the maximum or supercritical mass flow decreases along with recovery. In considering the two directions of translation, it should be pointed out that, for smaller movements of the spike, retraction of the compression surface permits stable regulation down to the engine-idle condition. Results for an angle of attack of 9° are also included in figure 7. These results are somewhat similar to those for the case of zero angle of attack; however, stability ranges, in general, have been decreased and larger θ_l must be used to attain the large stable subcritical ranges.

4006

Effect of Turbojet Engine on Inlet Stability

All the foregoing discussion has been concerned with results from cold-flow tests wherein a variable-area sonic exit was used to simulate the exit conditions anticipated in an actual engine application. Little information is currently available on the combined effects of an inlet operating in conjunction with a turbojet engine. Accordingly, at Mach numbers 1.8 and 2.0, a study was conducted on an annular nose inlet with a translating spike and a variable-bypass arrangement (see refs. 8 and 9). Performance was evaluated both with a cold-flow exit plug and with a J-34 turbojet engine. Results pertinent to the inlet stability ranges are shown in figure 8. Compared with the cold-flow plug, the engine had a definite stabilizing influence on subcritical operation of the inlet. The buzz regions are identified by the cross-hatched areas for the cold-flow plug and by the dotted portions of the figure for the engine. In all cases studied, the unstable regions were greater with the plug than with the engine. The actual damping mechanism, however, is not understood. Opening the bypass destabilized the inlet generally, but more so with the plug than with the engine. As buzz was initiated, the total-pressure amplitude at the compressor face was about the same in either case; however, the frequency of buzz with the engine was about twice that obtained with the plug.

These data are, of course, for a conservative engine which was choked at the exhaust nozzle and which was not designed for supersonic application. As such, these results should not be construed as being general. More advanced engines employing higher compressor blade loadings, with choking occurring at a much earlier station in the engine, may well yield considerably different results.

4006

CONCLUDING REMARKS

Supersonic inlet stabilization can be generally achieved for Mach numbers up to approximately 2.0 by careful consideration of the possible buzz-triggering conditions. Boundary-layer control and constant-area sections can be effectively utilized on inlets designed to provide stable flow regulation over the entire engine operating range. For Mach numbers above 2.0, the attainment of stability becomes increasingly more difficult as local Mach numbers (and thus normal-shock strengths) increase to aggravate further the shock-boundary-layer interaction problems.

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Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, November 1, 1955

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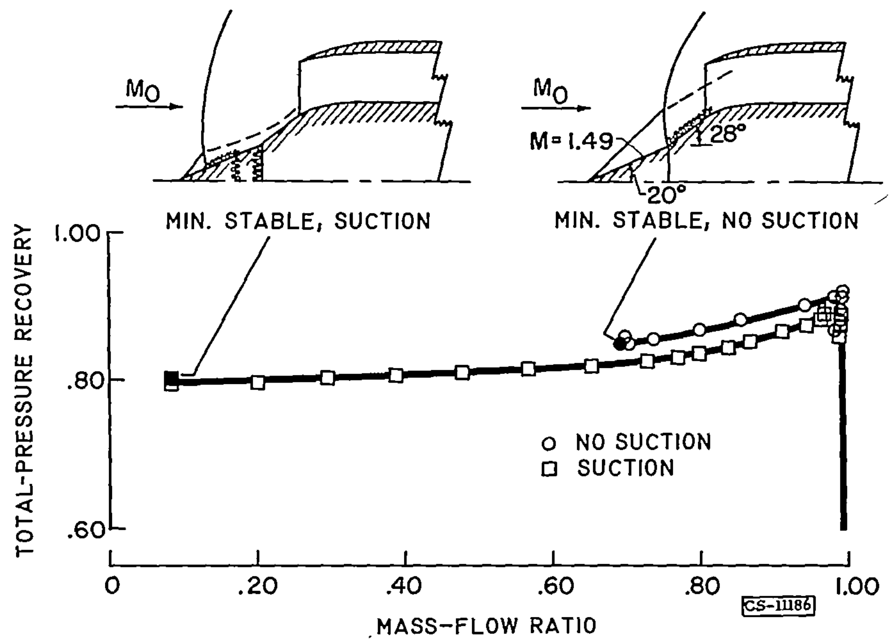


Figure 1. - Effect of boundary-layer suction. Free-stream Mach number, 1.9.

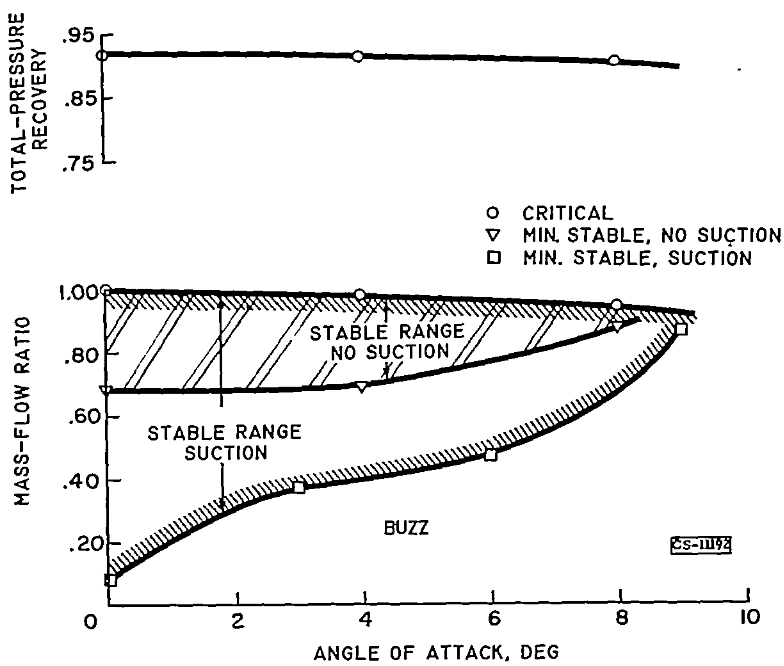
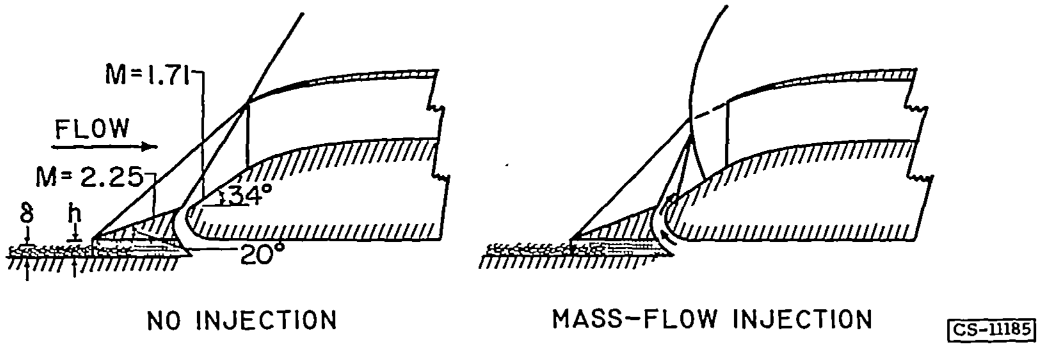


Figure 2. - Effect of angle of attack, 2-cone nose inlet. Free stream Mach number, 1.9.



h/δ	STABLE RANGE, NO INJECTION	STABLE RANGE, INJECTION	INJECTION MASS-FLOW RATIO, m_i/m_0	TOTAL-PRESSURE RECOVERY DECREMENT
1.05	$0(m_c)$	$0.24(m_c)$	0.02	0.04
.79	.03	.175	.03	.01
.26	.07	.49	.04	.002

Figure 3. - Effect of mass-flow injection. Free-stream Mach number, 2.96.

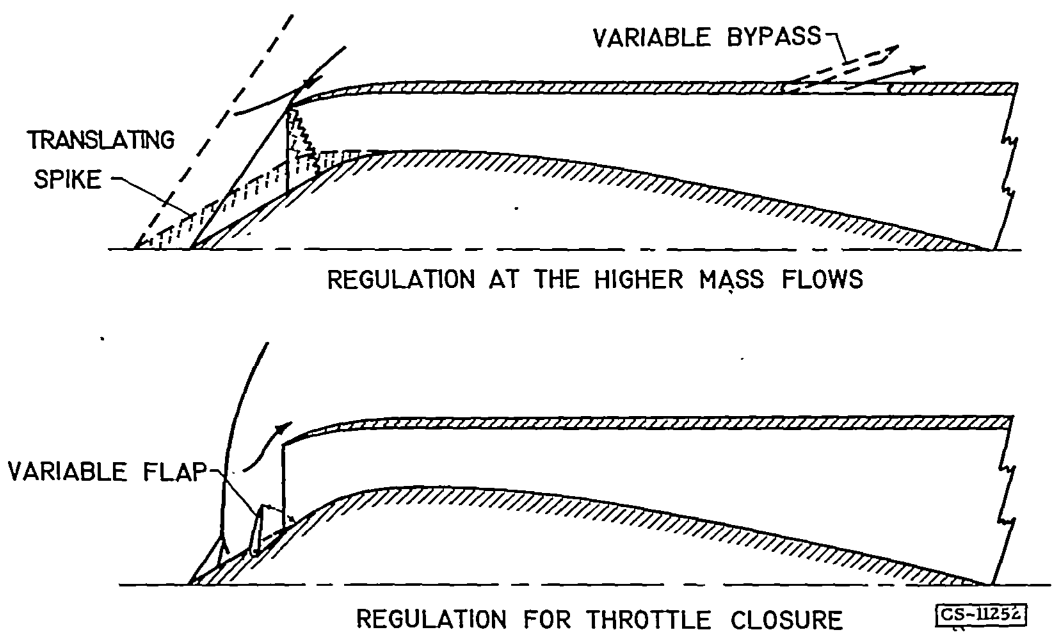


Figure 4. - Variable-geometry techniques for stable flow regulation.

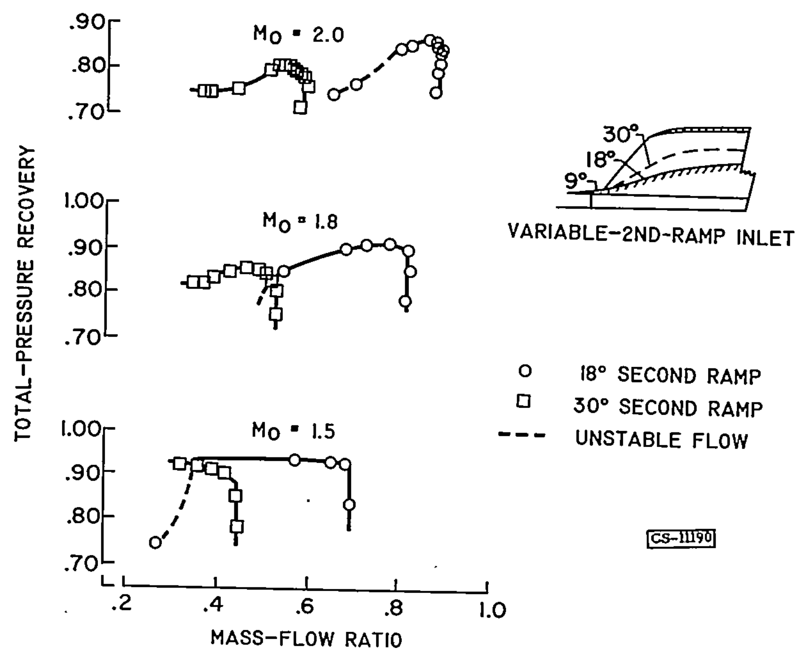


Figure 5. - Flow stabilization at low mass-flow ratios.

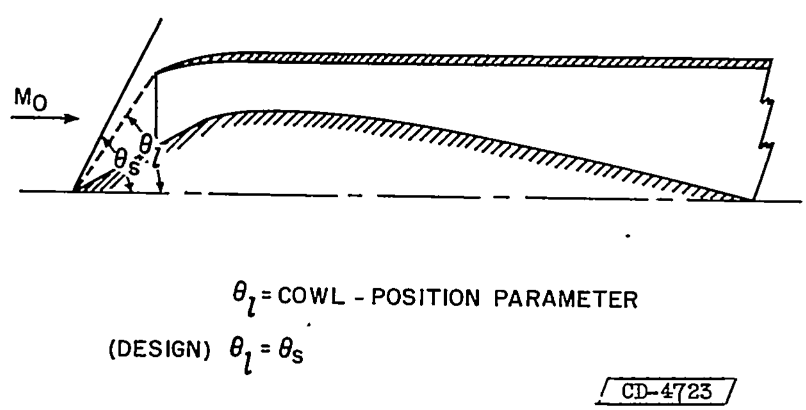


Figure 6. - Translating spike.

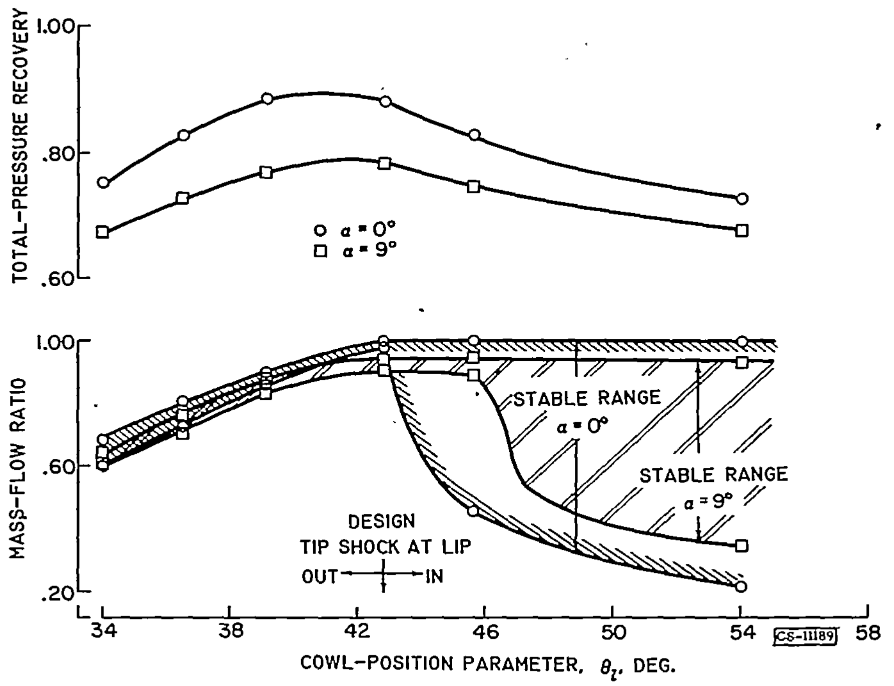


Figure 7. - Effect of spike translation. Free-stream Mach number, 2.0.

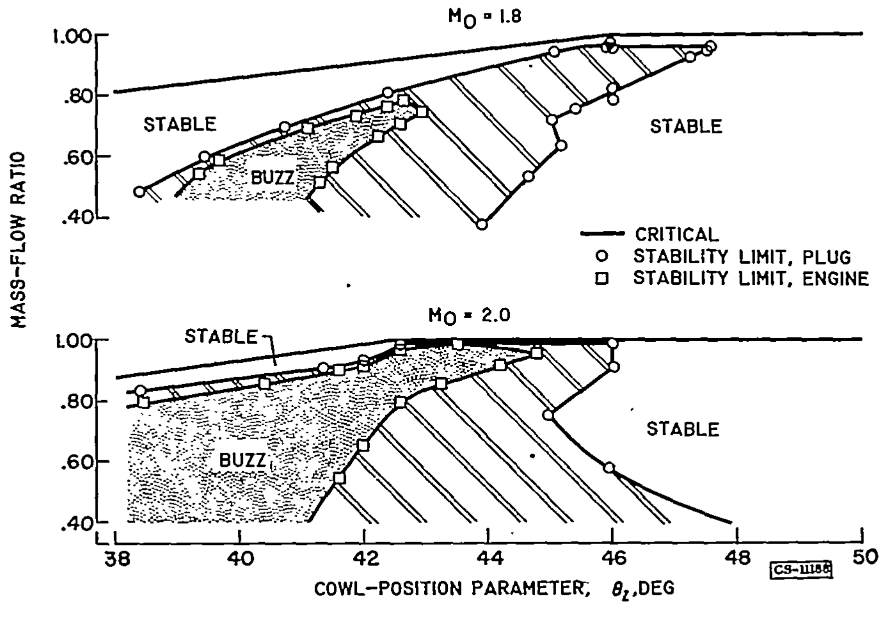


Figure 8. - Effect of the J-34 engine.

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