11. PERFORMANCE SUMMARY OF A TWO-DIMENSIONAL AND AN AXISYMMETRIC SUPERSONIC-INLET SYSTEM

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SUMMARY

The results of approximately three years of theoretical and experimental research effort are summarized for a large-scale two-dimensional and axisymmetric inlet system. A series of wind-tunnel tests have provided a background of experimental information showing the performance capabilities at Mach numbers from 0.6 to 3.2. The primary objectives were to investigate relatively short mixed-compression inlet systems with low external drag and to achieve high performance over the complete Mach number range with a minimum of engine-face distortion. The main conclusions are that the supersonic portion of the inlet system performed as predicted, and that the main difficulty in achieving high performance lay in the throat and subsonic diffuser. In addition, short subsonic diffusers appear practical when vortex generators are employed downstream of the throat.

INTRODUCTION

Air-induction system research and development programs at Ames Research Center have primarily supported the supersonic transport. The results, however, have a wide range of application to other supersonic aircraft. A series of wind-tunnel tests for large-scale two-dimensional and axisymmetric inlet systems have provided a background of experimental information showing the performance capabilities over most of the Mach number range up to 3.0. The present paper is intended as a brief summary of the results of approximately three years of theoretical and experimental research effort.

The programs were conceived to attain three major objectives. The first objective was to investigate mixed-compression inlet systems that were relatively short and had low external drag in an attempt to minimize the weight and drag while maintaining high internal performance. The second objective was to achieve high internal performance over the complete Mach number range, insuring maximum vehicle performance not only during cruise, but also during climb and acceleration. Because the engine-face distortion of short inlet systems is usually large, or unacceptable, the third objective was to investigate means for controlling the distortion to within acceptable levels.

Satisfactory performance of the supersonic transport has demanded propulsion systems with light-weight high-performance inlets. This has led to rather sophisticated designs which employ high internal contraction supersonic diffusers in combination with relatively short subsonic diffusers. Figure 1 shows the two-dimensional model mounted in the test section of the supersonic wind tunnel. It is designed for a Mach number of 3.0 and is capable of performing at off-design Mach numbers by opening the throat. The inlet is square, having a 14- by 14-inch capture area, and can be considered half of a complete inlet system feeding one engine. The external protuberances would not normally be present on an actual inlet and are peculiar to the ramp and boundary-layer-bleed control system of this model.

Figure 2 shows the axisymmetric model mounted in the test section of the transonic wind tunnel. It is also designed for Mach number 3.0. The model has a 20-inch capture diameter and is capable of performing at off-design Mach numbers by translating the centerbody. The small protuberances visible inside the inlet are vortex generators which have controlled the engine-face distortion. As mentioned for the two-dimensional model, the external protuberances are peculiar to this model and would not normally be present on an actual inlet. Both models are as large as practical for the test facility and can be considered 1/4 to 1/2 full scale. The models were tested in the wind tunnel primarily to determine the internal performance and bleed requirements from Mach number 0.6 to 3.2 at angles of attack and sideslip up to 8°. The Reynolds number per foot at Mach number 3.0 was about 2×10^{6} which corresponds to the Reynolds number per foot at 65,000 feet.

SYMBOLS

D capture diameter, axisymmetric inlet

m_{bl} bleed mass flow

m_{∞} free-stream mass flow

M_{∞} free-stream Mach number

P_{t_2} total pressure at engine face

$\bar{P}_{t_2}$ area weighted average total pressure at engine face

P_{t_∞} free-stream total pressure

α angle of attack

β angle of sideslip

TWO-DIMENSIONAL RESULTS

Satisfactory comparisons of two-dimensional and axisymmetric inlets are difficult to make, and as a consequence each inlet will be discussed separately. Figure 3 is a sketch of the two-dimensional model. The variable ramp assembly is positioned by adjusting the height of the throat ramp which can be differentially actuated for optimization of the area variation in the throat region. The angle of the initial ramp compression surface is fixed at 7°. The lower cowl surface can be translated aft, to the position indicated, for operation at Mach numbers lower than about 1.75. This translation provides for efficient external compression and spillage at low off-design Mach numbers. Perforated ramp and side-wall surfaces allow the boundary layer to bleed into three compartmented zones located above the ramps and then through ducts to controllable exit plugs. Bleed flows from the cowl and forward side wall are dumped directly to the outside airstream. Total pressure rakes are located at the simulated engine face. Vortex generators, which will be discussed subsequently, have been located in the throat region on the ramp and cowl surfaces, with eight on the cowl and eight on the ramp. The detailed design of the vortex generators follows the procedures of reference 1.

Initial tests were completed on the short subsonic diffuser, which represents an equivalent conical angle of about 9°. The maximum turning angle of the ramp surface was 25°. Design modifications, shown by the dashed lines, resulted in a subsonic diffuser which is equivalent to a 7° conical angle. With a slight increase in length, the ramp turning rates were reduced to a maximum of 16°.

Even though the off-design performance is important, the design performance is usually more important. Figure 4 indicates the supercritical performance at Mach number 3.0 and zero angle of attack and sideslip. Engine-face pressure recovery and distortion are plotted as a function of boundary-layer-bleed mass-flow ratio. The more usual abscissa for a supercritical plot is engine-face mass-flow ratio, which is merely the difference between 1 and the boundary-layer-bleed mass-flow ratio.

Initial tests of the two-dimensional inlet model utilizing the short subsonic diffuser indicated that flow along the ramp surface in the subsonic diffuser was separated. A total-pressure recovery of 87 percent was achieved with a bleed mass-flow ratio of about 13 percent with about 17-percent distortion. The modified system, as mentioned previously, had a subsonic diffuser with reduced turning angles on the ramp surface. The modification resulted in a peak pressure recovery of about 88 percent with about 14-percent bleed, but the distortion levels were high, 12 to 16 percent, because of flow separation on the subsonic ramp surface similar to that experienced with the shorter diffuser. Since a distortion of 10 percent is considered a nominal maximum, the forced mixing principle employing vortex generators was investigated for reducing the distortion. The test results show that adding four pairs of vortex generators across the width of the ramp side eliminated separated flow and reduced the distortion to 8 percent with a peak recovery of 90 percent. Placing vortex generators on both the cowl and ramp further reduced the distortion to 5 percent but increased mixing losses in the flow near the cowl surface, and resulted in a slight decrease in the maximum attainable recovery. The bleed mass-flow ratio for the bleed configuration used in this series of tests increased from about 11.5 percent to about 14.5 percent as a result of the terminal shock wave moving over the porous bleed surfaces in the throat. The dashed line is the envelope of peak performance for other bleed configurations. To avoid confusion the supercritical performance for all configurations was not plotted, but the trade between peak performance and bleed appears to be about a 1-percent increase in pressure recovery for 1-percent increase in boundary-layer bleed.

The effect of vortex generators on engine-face distortion is further clarified in figure 5. This slide presents total-pressure ratio profiles at the engine face, measured by a vertical center-line rake, for the peak recovery conditions of the previous figure. Without vortex generators the profiles for both the short and modified inlet are highly distorted. Adding vortex generators on the ramp side eliminated separation and improved the profile mostly near the ramp side, leaving the distortion near the cowl side much as it was. The addition of generators on the cowl side improved the profile to the point where the distortion was reduced to about 5 percent. The vortex generator tests were limited, and further improvements might be possible.

The performance at angles of attack and sideslip has shown some interesting results. Figure 6 shows the peak pressure recoveries at Mach numbers 3.0 and 2.5 for the short and modified inlets. As is expected the peak recovery at angle of attack is different from the recovery at angle of sideslip, the latter, shown by the dashed curves, having the lower recoveries. Two interesting results are to be noted. One is that the difference in recovery at angle of attack and sideslip is considerably greater at Mach number 3.0 than at 2.5. The other point is that the inlet with the short subsonic diffuser performs about as well as the one with the modified diffuser at Mach number 2.5, but is considerably poorer at Mach number 3.0 by 3 to 6 percent. This appears to be a result of the previously mentioned separated flow in the subsonic diffuser which persists or is aggravated with increase in angle of attack and sideslip. At Mach number 2.5 or less the flow is attached. This accounts for the better results shown for Mach number 2.5.

The final data plot for the two-dimensional inlet is shown in figure 7. Pressure recovery and distortion for peak operating conditions are plotted for a range of Mach numbers from 0.6 to 3.0. A break in the curves occurs at a Mach number of 1.3 because of test facility restrictions which limited boundary-layer bleed to about 2 percent at the transonic Mach numbers. At the higher Mach numbers from 1.55 to 3.0 the bleed varied from about 7 to 14 percent. Note that the cowl was retracted for Mach numbers below 1.75 and the inlet operated as an all external compression system, which probably accounts for the change in trend of the distortion in this Mach number range. The use of vortex generators on the ramp reduced distortion about 2 to 4 percent throughout the Mach number range without significantly affecting engine-face pressure recovery. A similar effect will be shown later for the axisymmetric inlet.

AXISYMMETRIC RESULTS

Figure 8 shows quarter section sketches of three of the inlet systems that have been tested. The upper quarter section shows two of the systems while the lower section shows the third system. In the upper quarter section one system is a rather long 1.86 diameter system measured from the lip to the engine face;

the other is a medium system 1.57 diameters long. Since the supersonic diffuser is the same for both of these systems, the subsonic diffuser accounts for the difference in length. The short inlet system in the lower quarter section is 1.40 diameters long. The upper and lower supersonic diffusers are almost the same length but differ in their theoretical efficiencies by about 3 percent. The upper inlet is capable of a theoretical throat recovery of 96.3 percent and represents an earlier design. Subsequent work revealed that a diffuser could be designed with 99.3 percent theoretical recovery if both the cowl contours and the forward portion of the centerbody were curved from 10° to 15° instead of being a straight 12.5° cone as in the earlier inlet mentioned above. The main problem in both supersonic diffusers has been to avoid boundary-layer separation due to shock-wave impingements. This requirement tended to establish a lower limit on inlet length of about that shown. Both supersonic diffusers were designed with the aid of a computer program (ref. 2) employing the method of characteristics. It has proved to describe accurately the flow field. An important consideration in a translating centerbody design is providing enough capture mass flow for the engine at off-design Mach numbers. These inlets supply about 40-percent mass-flow ratio at Mach number 1.0 and satisfy the demand of a selected turbofan engine. This type of inlet could satisfy a wide range of engines if a collapsible centerbody were employed.

The model has provision for removing the boundary layer through four porous bleed areas, two on the cowl and two on the centerbody. The porous areas are composed of holes drilled normal to the surface for a maximum porosity of about 40 percent. The porous pattern required by each inlet could be altered by filling some of the holes. Each of the four areas has a separate and controllable exit. The exits for the cowl bleed are illustrated in figure 8; those for the centerbody are further downstream and are not visible in the figure. By opening or closing these exits, the bleed mass flow could be varied for a given bleed configuration.

With these supersonic diffuser lines, three subsonic diffusers were designed. The basic area distributions follow a linear rate of change of Mach number from the end of the throat region to the engine face. The throat region which is two throat heights long has about a 2° divergence between the cowl and the centerbody. This method of design has proved effective in preventing boundary-layer separation in this portion of the inlet, something which was not accomplished in initial tests (1963) employing a linear area variation from the beginning of the throat region to the engine face. The equivalent conical angles of these diffusers (measured from the beginning of the throat region to the engine face) are about 11°, 15°, and 20° for the long, medium, and short inlet systems, respectively. Also shown in the figure are vortex generators located just downstream of the throat of the short and medium inlet systems. The long inlet was not tested with vortex generators. Tests of several generator heights showed that the optimum height was about 20 percent of the throat height. For the short inlet, 50 generators were located on the centerbody and 60 on the cowl. Additional details of the generators for the medium inlet are presented in reference 3.

Figure 9 compares the peak performance attained at various bleed mass-flow ratios for the three inlet systems just described at the design Mach number of 3.0 and at 0° angle of attack. For the initial tests a peak recovery of 86 percent with 13-percent bleed was attained with about 10-percent engine-face distortion. As mentioned previously, this inlet system employed a subsonic diffuser with a linear area variation. Since recoveries as high as 97 percent were measured in the throat, the main losses in performance lay in the subsonic diffuser. This diffuser had an initial rate of expansion that was too rapid causing separation and preventing attainment of higher recoveries. This led to the design shown for the long inlet whereby the initial rate of expansion was reduced preventing separation. This improved the performance to the point where a little over 90-percent recovery with about 11-percent bleed was attained. The engine-face distortion was about 10 percent and was the lowest distortion that could be attained with this diffuser which did not employ vortex generators. With the medium inlet employing vortex generators, not only was the distortion lowered to about 6 or 7 percent, but the level of recovery increased approximately 1 percent so that 91-percent recovery was attained with about 11-percent bleed. This latter result is attributable to the better distribution of the flow energy induced by the vortex generators. With the short inlet with the shortest subsonic diffuser but with a 3 percent more efficient supersonic diffuser, the recovery level increased about 2 percent over the bleed range indicated so that almost 93-precent recovery with 11-percent bleed resulted. However, the distortion level was about 3 percent higher than for the medium length inlet, probably because of the shorter subsonic diffuser.

Because the high levels of recovery shown for the short inlet have heretofore been unknown at a Mach number of 3.0, the supercritical performance was examined carefully and is shown in figure 10. Note that the scales for recovery and distortion have been reduced about 50 percent. Three levels of bleed flow are represented, an open, intermediate, and restricted bleed. The bleed through the two areas in the supersonic diffuser was held constant at 3.5 percent, and the total level of bleed was therefore regulated entirely by the adjustment of the two throat bleed exit settings. (The "unstart" angle of attack indicated on the recovery plot is explained in the next paragraph.) This inlet system is capable of recoveries of 93, 92, and 90 percent with bleed mass flow of about 12, 9.5, and 7.5 percent, respectively. Notice that limiting the throat bleed progressively limits the supercritical bleed mass-flow range from about 4 percent for the open bleed to about 1.5 percent for the restricted bleed. In addition, the distortion remains at or below about 10 percent over the useful supercritical range. The knee of the recovery curves represents the point where the terminal shock wave leaves the bleed area in the throat region and further movement downstream does not change the bleed mass flow. The distortion under these conditions can be quite high, 40 percent or greater.

Most axisymmetric mixed-compression inlets have been sensitive to small angle-of-attack changes caused by disturbances, such as gusts; that is, the inlets unstart easily with small changes in flow angle. This is not necessarily true for these inlets. Examples of the unstart sensitivity to flow angle at various operating points are shown on the curve for the intermediate bleed setting. With the shock wave in its most forward position, and without changing the inlet geometry, the inlet can be pitched only 1/2° before it unstarts, but if the performance is degraded 2 or 3 percent, as it would normally be for a practical operating condition, 2° or 3° of flow angle can be tolerated. This is with the inlet centerbody in the position for which the throat Mach number is about 1.2. With higher throat Mach numbers more tolerance is expected. It should be noted that the performance indicated at the operating points does not represent the performance at the unstart angles of attack. The performance is always something less at angle of attack. This is shown in figure 11 where comparable peak recoveries and the distortion are plotted as functions of flow angle for the three inlet systems. Each system exhibits its own peculiarities and a trend is not clear from the results. The short inlet, for instance, produces the highest recovery at 5° and 8°, a result which one might not expect. In addition, the distortion, even though it is high, is no worse than the medium length inlet at these same angles, and perhaps the generally lower distortion of the long inlet can be attributable to its longer length. The recovery for the short inlet, however, drops more rapidly from 0° to 2° than either of the other two systems sharing the same supersonic diffuser. This would seem to indicate the supersonic diffuser has an important influence on the angle-of-attack performance.

The three inlet systems have performed well over the complete Mach number range. Critical recovery and the accompanying distortion are shown in figure 12 to be comparable from Mach number 0.6 to 3.2. The results from Mach number 0.6 to 1.2 are optimum for a selected turbofan engine; that is, the net thrust minus the additive drag has been optimized to provide realistic as well as comparable results. The long and medium systems have shown about the same recovery capabilities but the distortion of the medium length inlet is generally lower. This indicates that the vortex generators are effective throughout the Mach number range even when (at other than the design Mach number) they are much displaced from the design position in which the cowl and centerbody generators are opposite one another. The short inlet shows a similar trend of recovery but at several percent increase at the higher Mach numbers. The distortion is generally lower than that of the long inlet. The peculiar dip in recovery at Mach number 2.9 for the short inlet is difficult to explain. This was not detected in the tests of the other two configurations but may also have been present. The points at $M_{\infty} = 3.2$ are overspeed conditions for the inlets; that is, the results at $M_{\infty} = 3.2$ were obtained with the Mach number 3.0 geometry settings, which accounts for the rapid deterioration of the distortion and recovery.

CONCLUDING REMARKS

The main conclusions to be drawn from the present summary for the two-dimensional and axisymmetric inlet systems are that the supersonic portion of the inlet systems performed as predicted, and that the main difficulty in achieving high performance lay in the proper design of the throat and subsonic diffuser. Also, short subsonic diffusers appear feasible when vortex generators are employed downstream of the throat. In addition, it appears that in order to achieve the highest performance a nearly isentropic supersonic diffuser must be employed which requires complete contouring of both the cowl and centerbody.

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TWO-DIMENSIONAL INLET MODEL

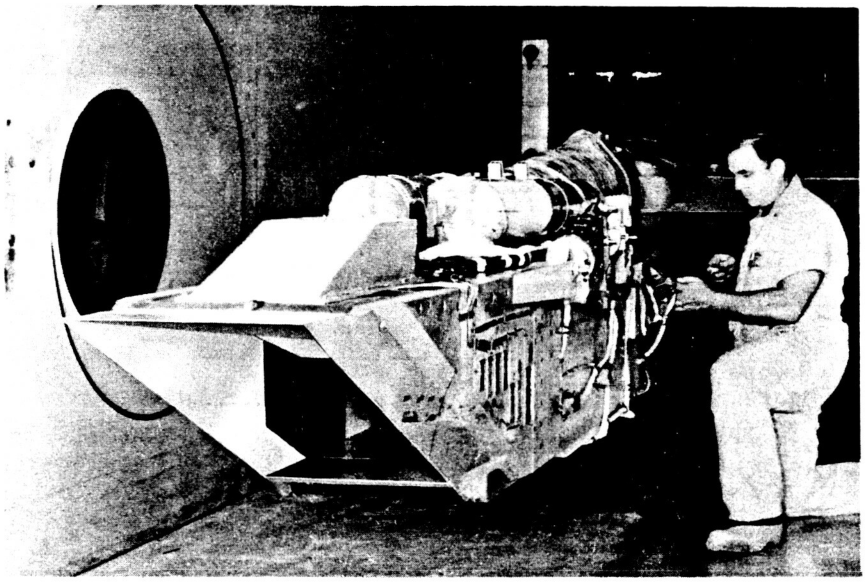

Figure 1 A-32238.1

AXISYMMETRIC INLET MODEL

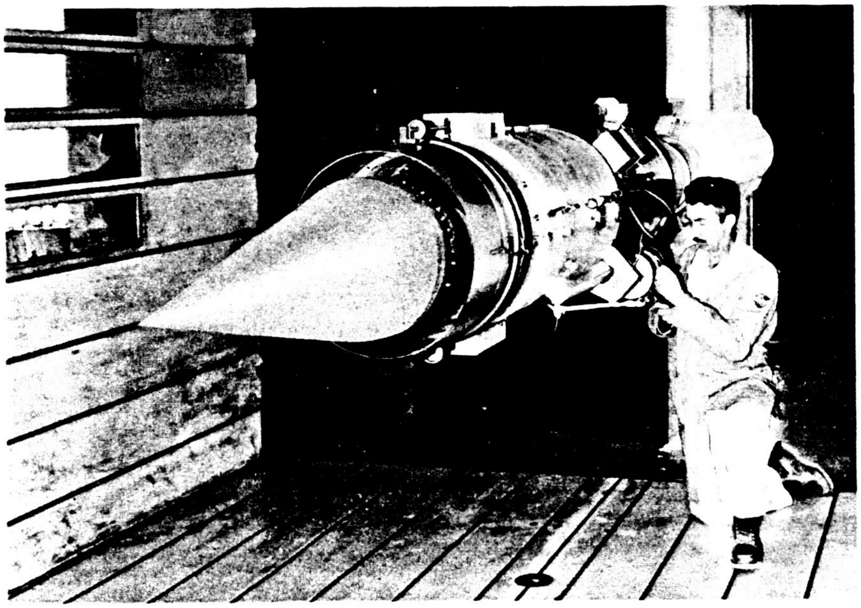

Figure 2 A-34092.1

TWO-DIMENSIONAL INLETS

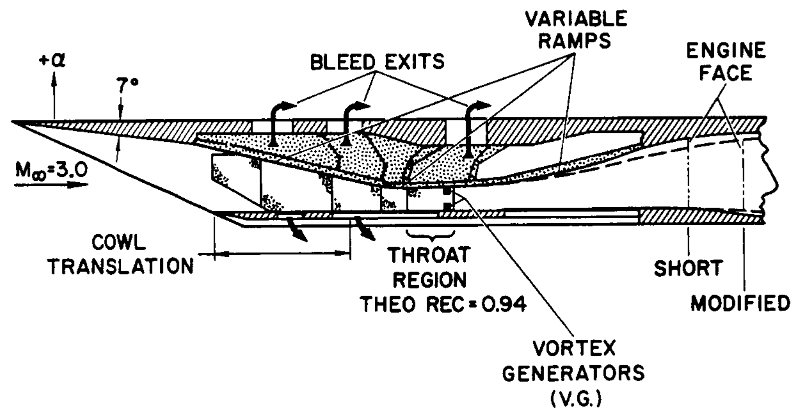

Figure 3

TWO-DIMENSIONAL INLETS, SUPERCRITICAL PERFORMANCE

$M_\infty = 3,\ \alpha = \beta = 0°$

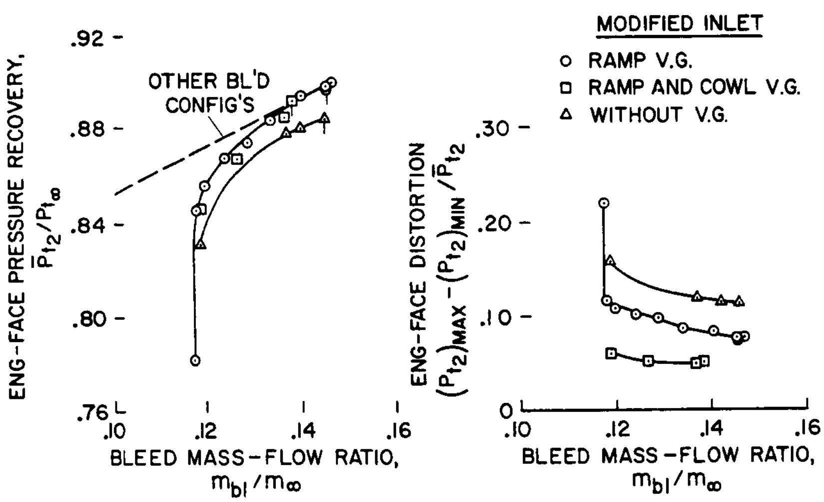

Figure 4

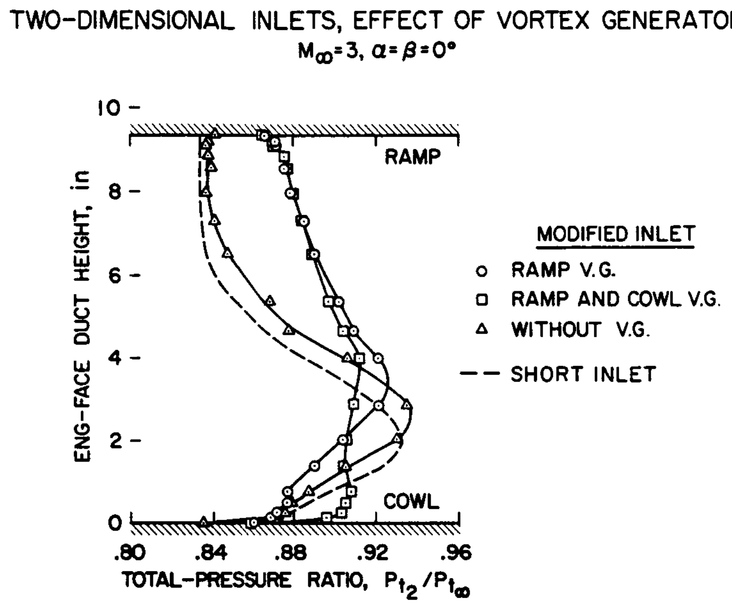

Figure 5

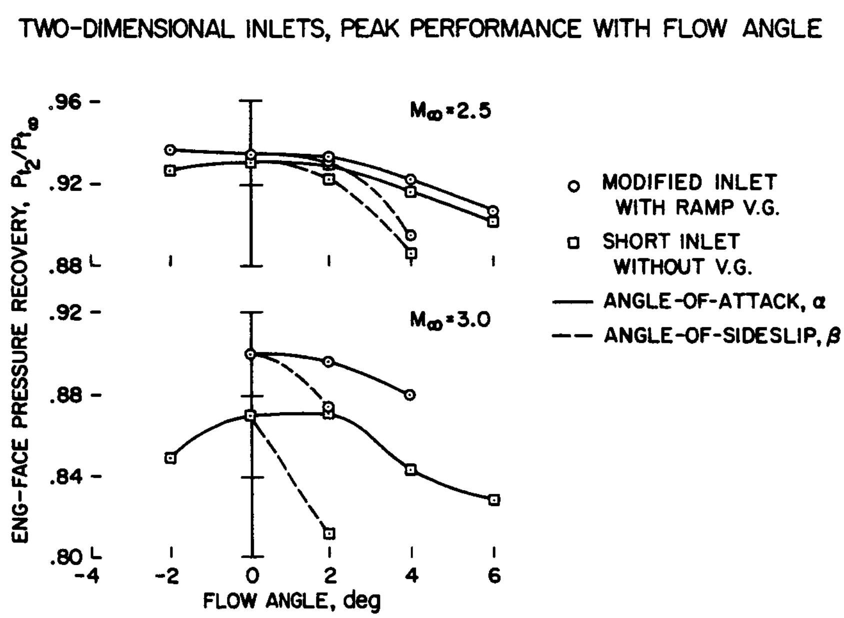

Figure 6

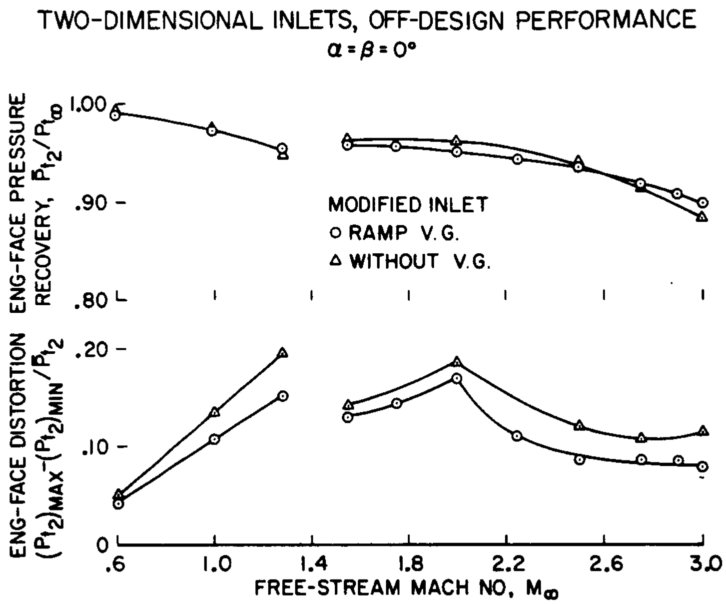

Figure 7

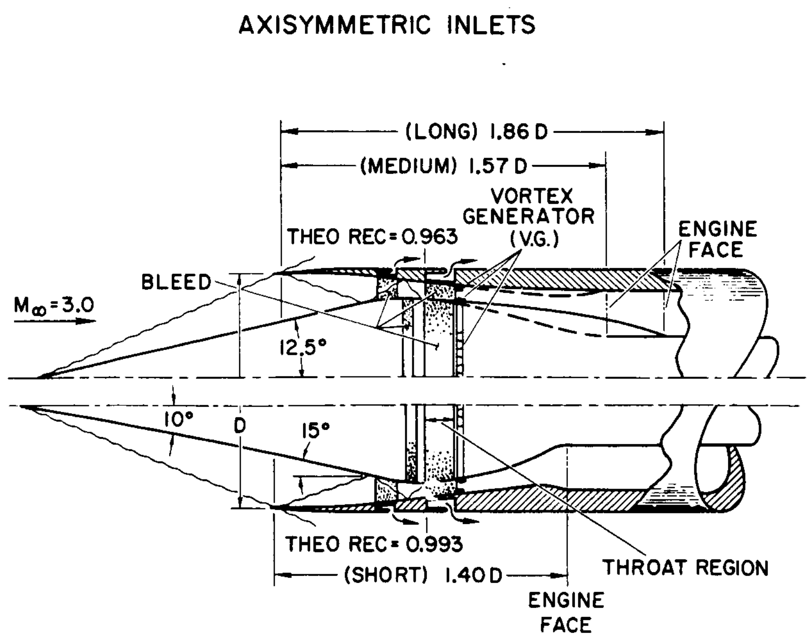

Figure 8

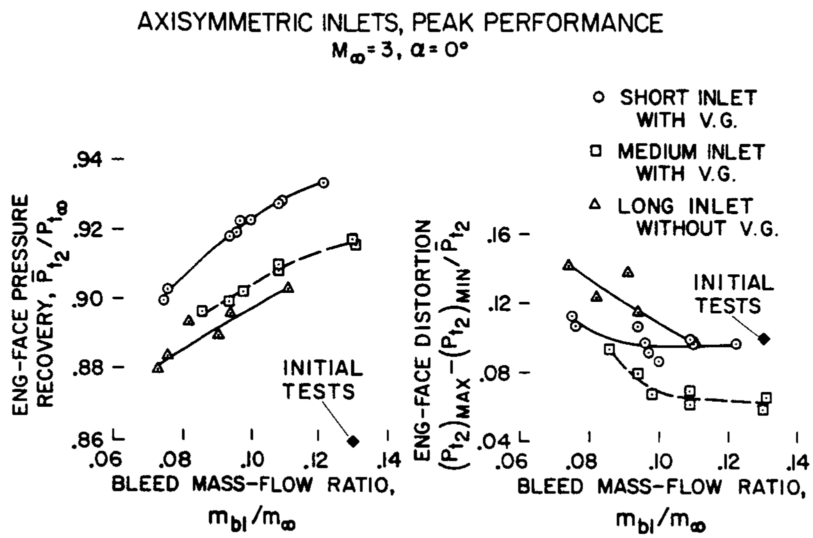

Figure 9

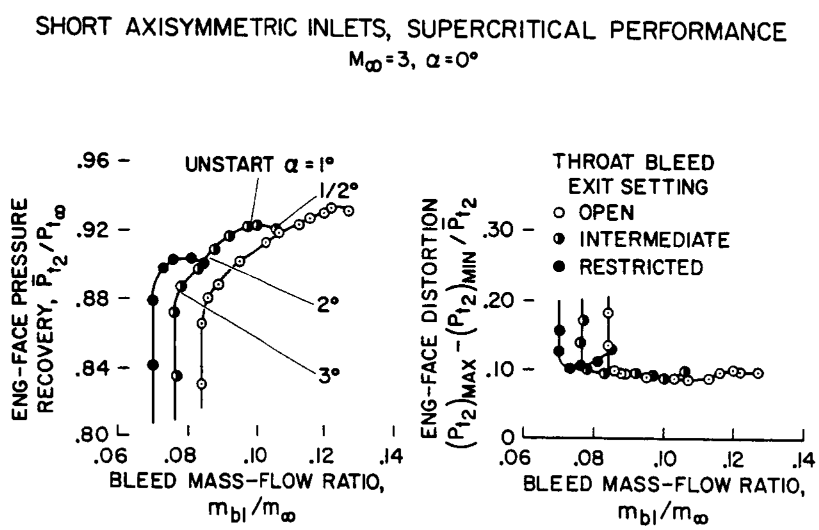

Figure 10

AXISYMMETRIC INLETS, PEAK PERFORMANCE WITH FLOW ANGLE

$M_\infty = 3$

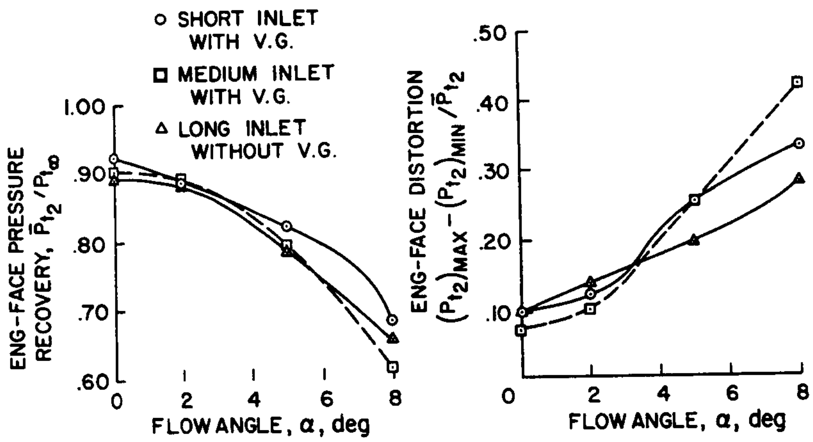

Figure 11

AXISYMMETRIC INLETS, OFF-DESIGN PERFORMANCE

$\alpha = 0$

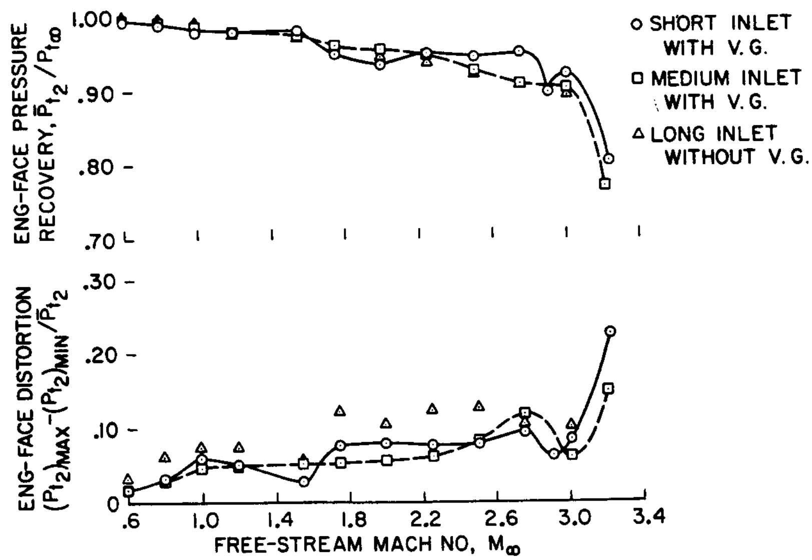

Figure 12