USSR, 1946; translated as NACA-TM-1399.

[6] Hayes, W. D., "Linearized Supersonic Flow," thesis, 1947, California Institute of Technology; reprinted as North American Aviation Rept. AL-222; available as Princeton University AMS Report 852.

[7] Hayes, W. D., "Pseudotransonic Similitude and First-Order Wave Structure," *Journal of the Aeronautical Sciences*, Vol. 21, 1954, pp. 721–730.

[8] Friedrichs, K. O., "Formation and Decay of Shock Waves," *Communications on Pure and Applied Mathematics*, Vol. 1, 1948, pp. 211–245.

[9] Whitham, G. B., "The Behavior of Supersonic Flow Past a Body of Revolution Far from the Axis," *Proceedings of the Royal Society* (London), Vol. A201, 1950, pp. 89–109.

[10] Whitham, G. B., "The Propagation of Weak Spherical Shocks in Stars," *Communications on Pure and Applied Mathematics*, Vol. 6, 1953, pp. 397–414.

[11] Whitham, G. B., "On the Propagation of Weak Shock Waves," *Journal of Fluid Mechanics*, Vol. 1, 1956, pp. 290–318.

[12] Lighthill, M. J., "Higher Approximations in Aerodynamic Theory," *General Theory of High Speed Aerodynamics*, edited by W. R. Sears, Sec. E, Vol. 6, pp. 345–489.

[13] Keller, J. B., "Geometrical Acoustics, I, The Theory of Weak Shock Waves," *Journal of Applied Physics*, Vol. 25, 1954, pp. 938–947.

[14] Busemann, A., "On the Relation Between Minimizing Drag and Noise at Supersonic Speeds," *Proceedings of the Conference on High-Speed Aeronautics*, edited by A. Ferri, 1955, pp. 133–144.

[15] Lomax, H., "The Wave Drag of Arbitrary Configurations in Linearized Flows as Determined by Areas and Forces in Oblique Planes," RM A55 A18, 1955, NACA.

[16] Guiraud J. P., "Acoustic géométrique, bruit ballistique des avions supersoniques et focalisation," *Journal de Mecanique*, Vol. 4, 1965, pp. 215–267.

[17] Resler, E. L., Jr., "Reduction of Sonic Boom Attributed to Lift," *Sonic Boom Research*, edited by I. R. Schwartz, SP-180, 1968, NASA, pp. 99–106.

[18] Resler, E. L., Jr., "Lifting Aerodynamic Configurations with No Sonic Boom," *AFOSR-UTIAS Symposium on Aerodynamic Noise*, 1968, to be published.

[19] McLean, F. E., "Some Nonasymptotic Effects on the Sonic Boom of Large Airplanes," TN D-2877, 1965, NASA.

[20] McLean, F. E., "Configuration Design for Specified Pressure Signature Characteristics," *Sonic Boom Research*, edited by I. R. Schwartz, SP-180, 1968, NASA, pp. 37–45.

[21] Jones, L. B., "Lower Bounds for Sonic Bangs." *Journal of the Royal Aeronautical Society*, Vol. 65, 1961, pp. 433–436.

[22] Jones, L. B., "Lower Bounds for Sonic Bangs in the Far Field," *Aeronautical Quarterly*, Vol. 18, 1967, pp. 1–21.

[23] Seebass, R., "Minimum Sonic Boom Shock Strengths and Overpressures," *Nature*, in press.

[24] Ferri, A., "A Report on Sonic Boom Studies," *Sonic Boom Research*, edited by I. R. Schwartz, SP-180, 1968, NASA, pp. 73–98.

MAY-JUNE 1969 J. AIRCRAFT VOL. 6, NO. 3

Study of a Family of Supersonic Inlet Systems

Norman E. Sorensen* and Donald B. Smeltzer*
NASA Ames Reasearch Center, Moffett Field, Calif.

AND

Robert W. Cubbison*
NASA Lewis Reasearch Center, Cleveland, Ohio

A theoretical and experimental program is reviewed showing the performance capabilities of a family of three large-scale axisymmetric inlet systems. The major objective of the program was to investigate relatively short inlet systems capable of high performance over the complete Mach number range. The tradeoff of important performance parameters such as engine-face pressure recovery and distortion for boundary-layer bleed was determined. The three inlets were designed for Mach numbers 2.5, 3.0, and 3.5. Vortex generators were used just downstream of the throats to control the flow distortion at the engine face. The Mach number 2.5 and 3.5 designs had engine airflow bypass systems. The results have shown that high performance was more difficult to achieve as the design Mach number increased, because the boundary layer occupied an increasing percentage of the flow in the throat. Also, the bypass systems have not presented serious performance problems.

Nomenclature

D = capture diameter
h = throat height
M = Mach number
m = mass flow
p_p = Pitot pressure
p_t = total pressure
$\bar{p}_t$ = area-weighted average total pressure
s = vortex generator height
α = angle of attack
δ = boundary-layer height

Subscripts

BL = bleed
BP = bypass
DES = design
max = maximum
min = minimum
u = unstart
2 = engine face
∞ = freestream

Presented as Paper 68-580 at the AIAA 4th Propulsion Joint Specialist Conference, Cleveland, Ohio, June 10–14, 1968; submitted June 10, 1968; revision received October 31, 1968.

* Research Scientists.

Introduction

STUDIES of supersonic cruise vehicles such as the supersonic transport have shown that the achievement of reasonable payloads is critically dependent on the achievement of low weight and high performance over the flight envelope. In particular, vehicle propulsion components such as the engine air inlet systems have been carefully studied. Low weight in these systems can be achieved by shortening them as much as possible. This has presented a challenging problem since short inlet system length is usually incompatible with the maintenance of high internal performance. Based on this problem, the performance of a family of axisymmetric inlet systems has been studied through experimental research programs conducted at Ames and Lewis Research Centers. The major objectives were to investigate theoretically and experimentally a family of three relatively short axisymmetric inlet systems designed to achieve high performance over the complete Mach number range; that is, high engine-face pressure recovery with low boundary-layer bleed requirements, low external drag, and low engine-face flow distortion were to be achieved over the Mach number range from zero to the design. Special attention was given to maintaining low engine-face flow distortion in the short inlet systems.

Inlet Descriptions

The three axisymmetric inlet systems tested are shown in Fig. 1. The inlets with capture diameters of about 20 in. can be considered about $\frac{1}{5}$ full scale for a supersonic transport. The models were large enough so that, with a wind-tunnel test pressure of 1 atm, natural boundary-layer transition occurred on the centerbody of each inlet upstream of the position of the first shock-wave impingement. Off-design operation of the inlets was accomplished through translation of the centerbody.† What constitutes a family for the three inlets is concerned mainly with the design of the supersonic diffusers. High external performance was built into each inlet by observing that low spillage and cowl drag can be achieved by having sufficiently low centerbody and initial cowl angles, respectively. The internal contours were developed through an iterative process with the aid of a computer program which employed the method of characteristics.[1] The program "tested" the contours by describing the flowfield and showed whether or not certain criteria for the internal design were met. That is, the length was as short as practical but not so short as to cause separation of the boundary layer.[2] In addition, each inlet was designed for a uniform Mach number, flow angle, and theoretical recovery in the throat.

The first inlet was designed for Mach number 2.5.[3] It had a 12.5° conical half-angle centerbody and a 0° initial internal cowl angle. With these angles, a relatively short supersonic diffuser was formed with the aid of the computer program. The supersonic diffuser was mated to a rather long subsonic diffuser with a linear area distribution from the end of the throat to the beginning of the bypass, and had an equivalent conical angle of 8°. The throat had 1° divergence between the cowl and centerbody contours, and the region from the beginning of the bypass to the engine face was essentially constant area. The total length of the system was 2.7 capture diameters measured from the cowl lip to the engine face. The inlet system could have been shorter if the subsonic diffuser were designed according to the procedures used for the next two inlet systems, but a proven design was the

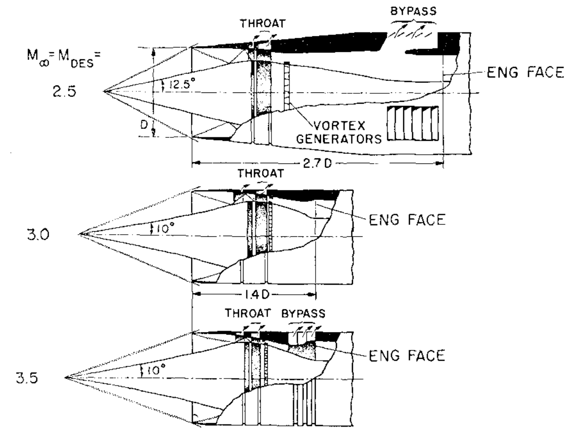

Fig. 1 Axisymmetric inlet systems.

criterion used in this case and it avoided the greater risk associated with a short subsonic diffuser. The second and third inlets were designed for Mach numbers 3.0[4] and 3.5, respectively. These inlets had centerbodies with initial conical half-angles of 10° which curved isentropically up to a maximum angle of 15°. The initial internal cowl angles were 0°. With these angles, supersonic diffusers were formed also with the aid of the computer program. These supersonic diffusers were mated to very short subsonic diffusers. The streamwise distribution of area in the subsonic diffusers was designed to produce a linear Mach number variation from the end of the throats to the engine-face stations. The equivalent conical diffusion angles from the beginning of the throats to the engine-face stations were about 20° and 28° for the Mach number 3.0 and 3.5 designs, respectively. The throat had about 2° divergence between the cowl and centerbody contours. The length of each inlet system from the cowl lip to the engine face was 1.4 capture diameters. Each inlet had four separate boundary-layer bleed areas, two on the cowl and two on the centerbody. The areas consisted of $\frac{1}{8}$-in. holes drilled normal to the surface and arranged to give a surface porosity of 40%. Each inlet had vortex generators[5] located just downstream of the throat region to control the engine-face airflow distortions. For the Mach number 2.5 design, vortex generators were used only on the centerbody, mainly to prevent separation of the flow from the centerbody side of the diffuser when a large percentage of the engine airflow was bypassed. The theoretical pressure recovery in the throat of each inlet ahead of the terminal shock-wave system was about 99% with a Mach number of about 1.25. The Mach number 2.5 design as well as the 3.5 design had an engine airflow bypass system located just ahead of the engine-face station. The Mach number 3.0 design, which was an earlier design, did not incorporate a bypass system. The bypass exit could be varied from closed to full open. The bypass flow for the Mach number 2.5 design passed through an annular slot into a plenum chamber and then through louvers to the freestream. For the Mach number 3.5 design, the bypass flow first passed through an annular surface consisting of $\frac{1}{8}$-in. holes drilled normal to the surface and arranged to give a surface porosity of 40%. The flow then passed to the freestream through a plenum chamber which was divided into three separate chambers to prevent recirculation of the bypass flow.

Performance

On-Design

For cruise vehicles, it is important that the design performance be high, since most of the flight time is spent at this condition. Figure 2 shows a comparison of the design super-

† If more engine-face mass flow is required at off-design Mach numbers to match a given turbojet or turbofan engine, a contracting instead of a translating centerbody would be required. However, the Mach number 3.0 and 3.5 designs with translating centerbodies will supply enough air over the complete Mach number range for two known engine designs.

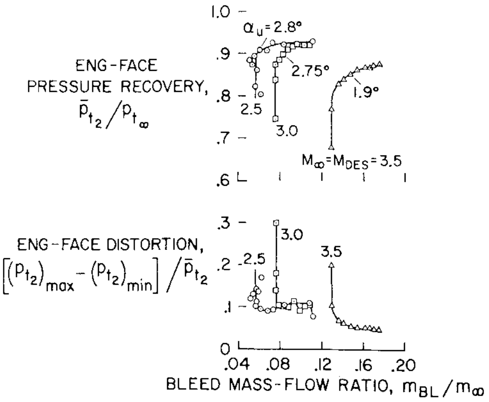

Fig. 2 Design performance at $\alpha = 0°$; no bypass.

critical performance for the family of three inlets without bypass airflow and at 0° angle of attack. These are results from an extensive experimental program where an optimum bleed configuration was determined as indicated in the previous figure. Engine-face pressure recovery and distortion are plotted as a function of bleed mass-flow ratio for each inlet at its design Mach number. Bleed mass-flow ratio is used as the independent variable instead of the more usual engine-face mass-flow ratio which, for the design operating condition as presented here, is 1 minus the bleed mass-flow ratio. The angles shown on the supercritical curves will be discussed shortly. As might be expected, it is more difficult to achieve high recovery with low bleed as the design Mach number increases. Ninety-percent recovery was attained for both the Mach number 2.5 and 3.0 inlets with about 5.5 and 8.5% bleed, respectively. Only about 85% recovery with 14.5% bleed was attained for the Mach number 3.5 design. However, for this design the distortion was only 5%‡ compared to about 10% for the two lower Mach number designs. Five-percent distortion was attained despite a subsonic diffuser with a high equivalent conical angle of 28° and a large area of porous bypass surface.

Unstart

For these inlets, sudden changes in angle of attack resulting from disturbances such as atmospheric gusts can cause the inlets to unstart.§ If the inlets were matched with engines so that the inlets operated very close to their maximum recovery, then only very small gustlike disturbances would cause the inlets to unstart. If, on the other hand, the inlets were matched so as to operate at pressure recoveries lower than the maximum, then it was found that the inlets remained started to appreciable angles of attack. Unstart angles of attack are shown on the pressure recovery curves at the indicated operating point. The angles indicate how high an angle of attack the inlets can tolerate before they unstart if no change in the inlet geometry, such as translation of the centerbody, is made. It can be seen that the Mach 3.5 design has the least tolerance, only 1.9°. The two lower Mach number designs have a relatively high tolerance of about 2.75°. More supercritical operation does not increase the unstart angles of attack, but less supercritical operation decreases the unstart angles.

Throat Profiles

Some insight as to why the bleed mass flow increases rapidly with the design Mach number and the pressure recovery decreases can be seen by examining the throat Pitot-pressure profiles presented in Fig. 3. Plotted is the Pitot-pressure recovery in the throat of each inlet as a function of the throat height at the design Mach numbers of 2.5, 3.0, and 3.5. As the design Mach number is increased for the same capture diameter the throat height decreases rapidly, but the boundary-layer thickness does not decrease proportionately, with the net result that the low-energy boundary layer occupies a much larger percentage of the throat area at the high Mach numbers. Because of this, more boundary-layer bleed has been required to maintain high-pressure recovery as the design Mach number increases.

Vortex Generators

The use of vortex generators in the subsonic diffusers of the two higher Mach number inlet designs have been found to be essential for attaining high performance. An indication of the optimum vortex generator height and the relative effectiveness of these devices at Mach number 3.0 is shown in Fig. 4. Once again, engine-face pressure recovery and distortion are plotted as a function of bleed mass-flow ratio.¶ Four different ratios of vortex generator height to bondary-layer height are represented.** The optimum vortex generator height appears to be about 120% of the boundary-layer height. Not only is the distortion the lowest (about 6 to 7%), but the recovery was the highest. This latter result was attributable to a more optimum distribution of the area-weighted flow energy at the engine face resulting from mixing the low-energy boundary layer with the high-energy core flow. Without vortex generators the distortion is more than double that for the optimum generator height ratio. In addition, the recovery was about 2 to 3% less than for the optimum.

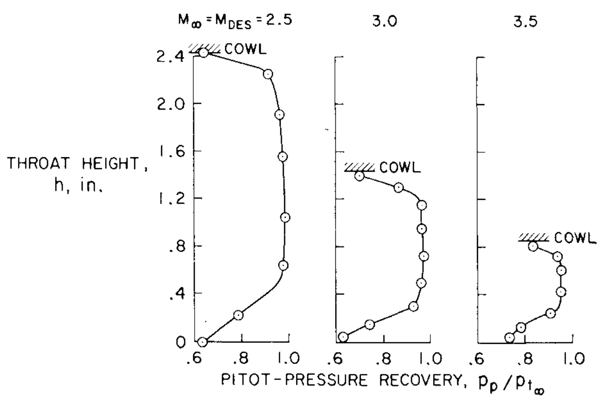

Fig. 3 Throat flow profiles at $\alpha = 0°$.

‡ This low distortion was perhaps easier to attain for the Mach number 3.5 design because the engine-face Mach number was lower than for the two lower Mach number designs. That is, the lower the average engine-face Mach number, the closer the static pressure is to the total pressure, which tends to reduce the possibility of high distortion.

§ These gustlike disturbances cannot be detected by automatic control systems in time to permit sufficient translation of the centerbody to keep the inlets started as would be done during an aircraft maneuver. It is, therefore, important to determine the maximum angle of attack for which the inlets remain started without translation of the centerbody as a function of the pressure recovery at 0° angle of attack.

¶ The data are for an inlet system slightly different from the one previously compared, but the results are believed to be generally applicable for inlet systems with similar short subsonic diffusers.

** The spacing between generators should also be considered. For the present case the spacing is about three generator heights. This is believed to be close enough so that for the broadest spacing, vortex flow induced by the generators dissipates enough to produce uniform flow by the time the flow reaches the engine face.

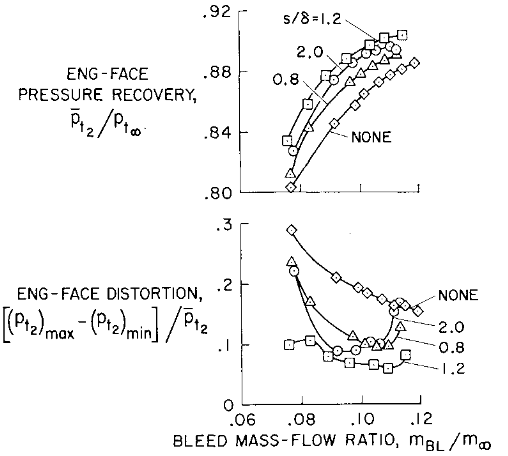

Fig. 4 Optimum vortex generator height at $M_\infty = 3.0$; $\alpha = 0°$.

Angle of Attack

Each of the three inlet systems has been tested at angle of attack. Performance has varied considerably from inlet to inlet and is shown in Fig. 5. Maximum engine-face pressure recovery and the associated distortion is plotted as a function of angle of attack. As expected, the recovery for all three inlets decreased and the distortion increased with angle of attack. However, the recovery decreased more rapidly with angle of attack as the design Mach number increased. The distortion has shown a different trend. The inlet with the lowest distortion was the Mach number 3.5 design whereas the two lower Mach number designs showed a higher distortion level, and there was little difference between them. Distortion at 8° angle of attack was about 35 to 40% for the Mach number 3.0 and 3.5 designs. Why such variations take place is not well understood, but it is associated with the unequal compression in the supersonic diffuser aggravated by flow separation in the subsonic diffuser.

Bypass

Concern has been expressed regarding the possible deleterious effects on pressure recovery and distortion that might result from locating the bypass systems as near to the engine face as in the Mach number 2.5 and 3.5 designs. However, this apparently has not been a problem for these inlets at the design condition as can be seen in Fig. 6. Maximum engine-face pressure recovery and the associated distortion are plotted for a wide range of bypass mass-flow ratio at 0° angle of attack for both inlets at their respective design Mach

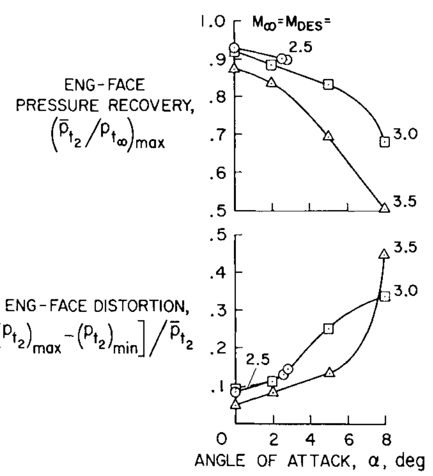

Fig. 5 Performance at angle of attack; no bypass.

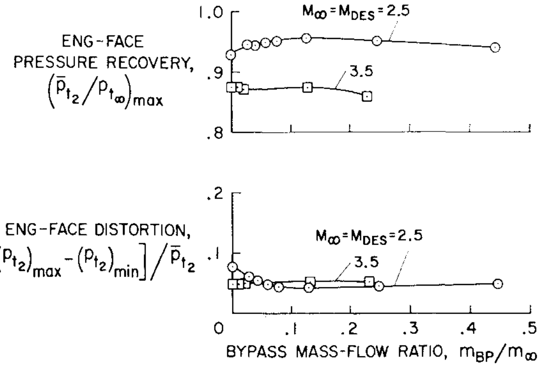

Fig. 6 Performance with bypass at $\alpha = 0°$.

numbers of 2.5 and 3.5. Although only a small range of bypass mass flow is considered important with a started inlet, the wide range of bypass shown can be considered important for restart conditions. Increasing the bypass mass-flow ratio to 10% for the Mach number 2.5 design had a favorable effect increasing the pressure recovery and decreasing the distortion. Further increase in the bypass mass flow did not have an appreciable effect on either recovery or distortion. For the Mach number 3.5 design, the pressure recovery and distortion varied only slightly over the bypass mass-flow ratio shown.

Off-Design

The off-design condition rather than the design is where the effects of bypass mass flow on the performance appears to be significant at least for the Mach number 3.5 design. For this inlet, Fig. 7 shows a plot of the same parameters as in the previous figure for Mach numbers from 3.5 to 2.0. The greatest variation in pressure recovery and distortion with bypass mass flow occurred at the lower Mach numbers. The pressure recovery decrements at Mach numbers 2.0 and 2.5 were about 4% at a bypass mass-flow ratio of about 25%. Maximum distortions of 12 and 14% which most engines can tolerate for a short time occurred at a bypass mass-flow ratio of about 20% for Mach numbers 2.0 and 2.5, respectively. At Mach numbers 3.0 and 3.5, little change in pressure recovery and distortion is evident over the entire range of bypass mass-flow ratios shown.

As previously mentioned, it is important to maintain high performance off-design as well as at the design Mach number. Figure 8 shows off-design performance for the three inlets over a wide range of Mach numbers for 0° angle of attack and with no bypass flow. Plotted are maximum engine-face pressure recovery and the associated distortion as a function of freestream Mach number. For the Mach number 2.5 inlet design, data are currently available down to only Mach number 2.0. For this design the recovery is about 2 to 3%

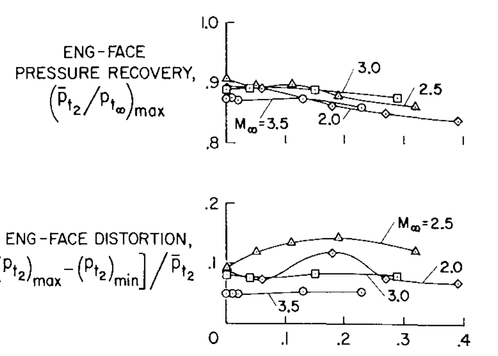

Fig. 7 Off-design performance with bypass at $\alpha = 0°$; $M_{DES} = 3.5$.

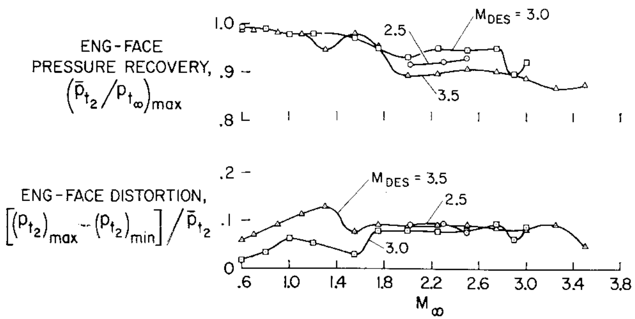

Fig. 8 Off-design performance at α = 0°; no bypass.

lower than for the Mach number 3.0 design, but it should be remembered that the recovery was attained with a low boundary-layer bleed mass-flow ratio of about 5.5%. The recovery was generally high over the complete Mach number range for the Mach number 3.0 design. The sudden dip in recovery at Mach number 2.9 is not yet well understood, but it is believed that local separation in the supersonic diffuser caused by inadequate boundary-layer control is responsible. For the Mach number 3.5 design, the recovery has not been as high as might be possible when compared to the Mach number 3.0 design. The comparatively low level of recovery down to Mach number 2.0 is believed to be caused by local separation on the centerbody of the supersonic diffuser and separation of the flow in the bypass area of the cowl. When these separations are not present, the recovery suddenly jumps to the Mach number 3.0 inlet values at Mach numbers 1.55 and 1.75 and 0.6 to 1.1. The drop in recovery at Mach number 1.3 is attributable to separation of the flow from the cowl in the subsonic diffuser. So far as distortion is concerned the Mach number 3.5 inlet has not performed as well as the 3.0 inlet in the transonic region, where the distortions have been double or more the distortions of the Mach number 3.0 design. The distortion at supersonic Mach numbers has been generally slightly higher. It is clear that use of vortex generators does not guarantee low distortion as evidenced in the transonic region, probably because the generator position and geometry are dictated by the requirements for operation at the design Mach number condition. It is believed that if the Mach number 3.5 inlet system is redesigned so that high-pressure gradients are avoided in the supersonic diffuser and rapid turning is avoided in the subsonic diffuser, the performance can be brought up to the level shown for the Mach number 3.0 inlet system.

Concluding Remarks

It has been shown that it is more difficult to achieve high on-design pressure recovery with low boundary-layer bleed as the design Mach number increases because an increasing percentage of the throat flow is occupied by the boundary layer. There appears to be an optimum on-design vortex generator height. Distortion without generators has been more than double that with the optimum generators. However, the use of generators does not guarantee low distortion at off-design Mach numbers. Bypassing large quantities of flow just ahead of the engine face in a very short subsonic diffuser has not caused undue performance penalties at the maximum performance operating condition.

References

[1] Sorensen, V. L., "Computer Program for Calculating Flow Fields in Supersonic Inlets," TN D-2897, 1965, NASA.

[2] Kuehn, D. M., "Experimental Investigation of the Pressure Rise Required for the Incipient Separation of Turbulent Boundary Layers in Two-Dimensional Supersonic Flow," Memo 1-21-59A, 1959, NASA.

[3] Cubbison, R. W., Meleason, E. T., and Johnson, D. F., "Effects of Porous Bleed in a High Performance Axisymmetric, Mixed-Compression Inlet at Mach 2.50," TM X-1692, 1968, NASA.

[4] Smeltzer, D. B. and Sorensen, N. E., "Investigation of a Nearly Isentropic Mixed-Compression Axisymmetric Inlet System at Mach Numbers 0.6 to 3.2," TN D-4557, 1968, NASA.

[5] Taylor, H. D., "Summary Report on Vortex Generators," Rept. R-05280-9, March 7, 1950, Research Dept., United Aircraft Corp.