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Advanced Supersonic Inlet Technology

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Recently, relatively new analytical procedures have been successfully used to design bleed systems for mixed-compression inlets designed to operate efficiently up to Mach number 2.65. The procedures used constitute a major advance in inlet technology by offering a promising approach to attain high internal and external performance for mixed-compression inlets that operate over a large supersonic Mach number range. Unfortunately, there is a lack of data describing bleed hole performance characteristics to verify these procedures at high Mach numbers. Further, as the Mach number increases, inlets with bleed systems become more difficult to design because the available methods do not properly account for the boundary-layer growth in large adverse pressure gradients, particularly in shock-wave impingement regions. Also, regardless of the analytical methods available, the design of bleed systems that can operate efficiently over wide Mach number ranges is a challenging problem because it is difficult to satisfy both cruise and off-design bleed requirements. This paper briefly discusses the analytical procedures for designing advanced inlet systems and suggests facility modifications wherein the procedures can be verified on large-scale inlet models up to approximately Mach number 4.5.

Nomenclature

A_c = capture area
d = hole diameter
h = throat height
l = hole length
M = Mach number
$\bar{M}_L$ = average local Mach number
m = mass flow
m_∞ = capture mass flow, $\rho_\infty U_\infty A_c$
N = boundary-layer power-law exponent
p = static pressure
p_p = pitot pressure
p_t = total pressure
$\bar{p}_t$ = area-weighted average total pressure
p_{p_L} = plenum chamber pressure
Δp = pressure rise to throat of inlet
R = cowl lip radius
U = velocity
x = axial station measured from tip of centerbody
α = angle of attack
β = angle of sideslip
δ^* = boundary-layer displacement height
η_{KE} = kinetic energy efficiency
ρ = density

Subscripts

b_l = bleed
DES = design
L = local
e = freestream exit
2 = engine face
δ = boundary-layer edge
∞ = freestream

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Introduction

MANY aircraft have been developed that operate efficiently up to approximately Mach number 2.5. Between Mach number 2.5 and 3.5, only a few aircraft have been developed and beyond Mach number 3.5, virtually none. Propulsion research for such aircraft have, unfortunately, followed a similar pattern; i.e., little effort has been placed on analytical and experimental research in the Mach number range above 2.5. As a result, the technology for the design of efficient inlet systems for aircraft flying below Mach number 2.5 is well developed. Conversely, in the higher supersonic range (from approximately $M = 3.0$ to 4.5), current technology does not adequately account for such phenomena as high adverse pressure gradients, severe shock-wave boundary-layer interactions, and the greater Mach number range over which the inlet systems must operate efficiently. However, it appears from recent studies that the required technology can be developed. This paper will show the trend of advanced supersonic inlet technology and some of the important analytical and ex-

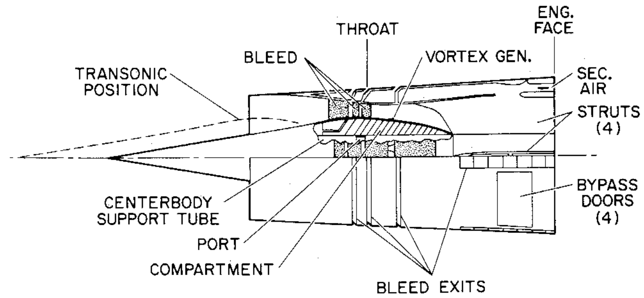

Fig. 1 Axisymmetric inlet system; $M_\infty = 2.65$, $\alpha = 0°$.

perimental requirements for further development of this technology.

Technology Base

Recent inlet designs that have advanced the base of supersonic inlet technology are axisymmetric mixed-compression inlets designed for high internal and external performance up to the cruise Mach number of 2.65.[1,2] Figure 1 shows details of one of the systems. The inlet was designed for shock on lip at $M = 2.65$ and isentropic compression to $M = 1.25$ in the throat. The inlet operated in the "started" mixed-compression mode above Mach number 1.6 and in the external compression mode below 1.6. Transonically, the centerbody was extended as shown and provided nearly the maximum transonic mass flow possible for a Mach 2.65 axisymmetric inlet system.[3]

The contours for the supersonic diffuser were designed using the conventional method of characteristics[4] and they were not adjusted to account for the boundary layer. Results from previous tests for similar inlets designed for $M \simeq 3.0$ and below indicate that the usual amounts of bleed flow required to prevent boundary-layer separation closely compensated for the blockage effects of the boundary layer throughout the supersonic diffuser.[5] For the subsonic diffuser, a trumpet-shaped contour provided a nearly linear variation of Mach number from the throat to the skirt of the centerbody just upstream of the engine face station. Previous tests[5] show that very short and efficient subsonic diffusers that provide low engine-face flow distortion can be designed this way if vortex generators are located downstream of the throat (Fig. 1).

The shaded areas (Fig. 1) show where rows of boundary-layer bleed holes were located. The holes in the supersonic diffuser were slanted forward, and those in the throat were normal to the surface. The bleed hole plenum chambers were compartmented to prevent the high-pressure bleed airflow from recirculating into the low-pressure upstream regions, since without compartmentation the recirculation would have resulted in separation of the boundary layer. Slanting the holes and compartmenting the plenums allowed higher pressure recoveries in each plenum chamber, minimizing the momentum drag of the overboard bleed flow. The diameter of all the bleed holes, approximately equal to the local boundary-layer displacement thickness ($d \simeq 0.159$ cm), was believed to be small enough to avoid undue disturbance to the boundary layer. The design features of this inlet required a unique boundary-layer bleed system for the translating centerbody. For good performance at supersonic Mach numbers >1.6, boundary-layer removal from the centerbody surface near the inlet throat (minimum annulus area) was required. With this inlet design, the inlet throat remained fixed relative to the cowl as the centerbody was extended from the $M = 2.65$ design position. Consequently, the location of the inlet throat relative to the centerbody and the regions on the centerbody requiring boundary-layer bleed changed as the centerbody was extended. To accomplish changes in location of the required bleed, a unique traveling boundary-layer bleed system was devised.[1,2] A number of compartments within the centerbody (Fig. 1) were arranged so that each provided a connection between the boundary-layer removal holes on the surface of the centerbody and the passages within the support tube through a valving arrangement, much like a conventional sleeve valve. Proper arrangement of the receiving ports in the centerbody support tube and the discharge ports from the centerbody compartments provided the desired changes in location of boundary-layer removal from the centerbody surface as the centerbody was extended.

The inlet shown in Fig. 1 was tested initially at ¼-scale,[3] and a bleed system was developed in the wind tunnel using cut-and-try methods. However, more tests were conducted later at ⅓-scale,[1] but this time with a bleed system designed using new analytical procedures. These procedures constitute one of the major advances in inlet technology and promise to save many wind-tunnel testing hours by avoiding much of the usual cut-and-try wind-tunnel bleed system development.[6]

Analytical Procedure

The analytical procedure of Ref. 1 allows prediction of the required bleed system performance before testing and permits rapid optimization of inlet performance over the complete started supersonic Mach number range of operation. A simple example of how the procedures are used is indicated in Fig. 2. The key to locating bleed holes lies in predicting where the boundary layer will separate. The criterion used for preventing boundary-layer separation is based on maintaining limits on the exponent N in the familiar boundary-layer velocity-profile equation shown in Fig. 2. Experience has shown that to prevent separation, N should be approximately 3 or greater[1] in the supersonic diffuser upstream of the throat station and approximately 7 or greater[1] in the throat region where the terminal shock wave is positioned for started operation. Thus, for the typical centerbody contour shown in the figure, the boundary-layer exponent distribution is first calculated for the inlet at the cruise Mach number with no boundary-layer removal using the calculated inviscid pressure distribution or Mach number distribution as an input for the boundary-layer computer program.[7] For the simple example shown, N becomes <3 between the shock-wave impingements ($x/R \simeq 3.8$) and tends sharply to zero just before the throat region ($x/R \simeq 4.2$). To prevent separation, part of the boundary layer is uniformly removed in the regions shown in the figure by inputing a specified amount of bleed flow (i.e., $m_{bl}/m_\infty \simeq 0.02$) into the boundary-layer computer program. Note that when mass is removed through a bleed area, the velocity distribution

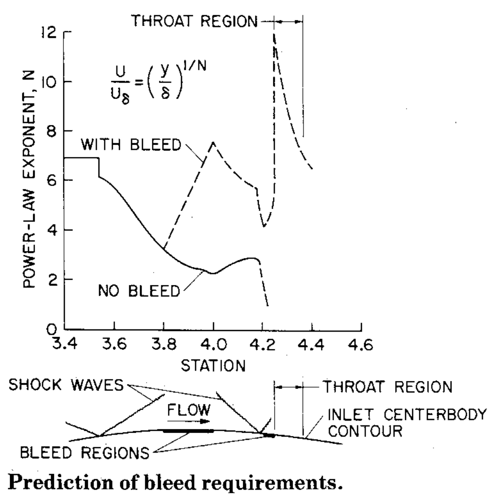

Fig. 2 Prediction of bleed requirements.

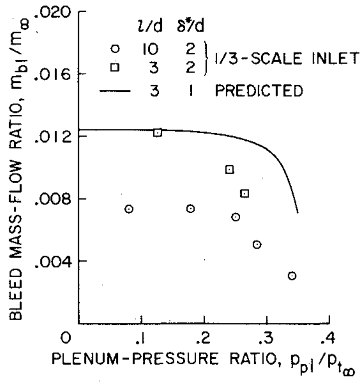

Fig. 3 20° bleed-hole characteristics; M_L = 1.37.

through the boundary layer at the beginning of the bleed area is changed from its predicted distribution—one that is near separation—to a more fully turbulent profile at the end of the bleed area. The profile established depends, of course, on the amount of mass removed through the bleed area; i.e., immediately upstream of the bleed region the boundary-layer equations are solved subject to the wall boundary conditions where the velocity on the wall is zero and there is no flow through the wall. In the bleed region, the velocity through the wall is not zero, but is changed to a prescribed function of x. At the end of the bleed area, the original wall conditions are again imposed. Thus, various values of m_{bl}/m_∞ are tried in the bleed area until just enough boundary layer is removed to prevent separation anywhere along the surface. Figure 2 shows that with bleed (dashed curve) $N > 3$ between the shock-wave impingements and >7 in the throat, meeting the separation criteria. For an actual inlet system (Fig. 1), N distributions are calculated in the aforementioned manner over the complete started Mach number range to see if the bleed system can meet separation requirements ($N > 3$) at all started Mach numbers. If not, the bleed hole arrangement is modified until a satisfactory distribution of N is obtained at all started Mach numbers. This analytical procedure is used instead of wind-tunnel experimental procedures wherein bleed holes are opened and closed in a cut-and-try fashion until the inlet performance parameters of high engine-face pressure recovery, acceptable distortion, and low bleed mass flow are achieved.

Once the required flow through the bleed regions is established, bleed hole dimensions and patterns that will remove the required flow are selected. This selection requires prediction of the mass flow that can be removed from various bleed holes. Bleed hole performance is based on experimental data where some of the significant geometric parameters are l/d, δ^*/d, and hole slant angle. In

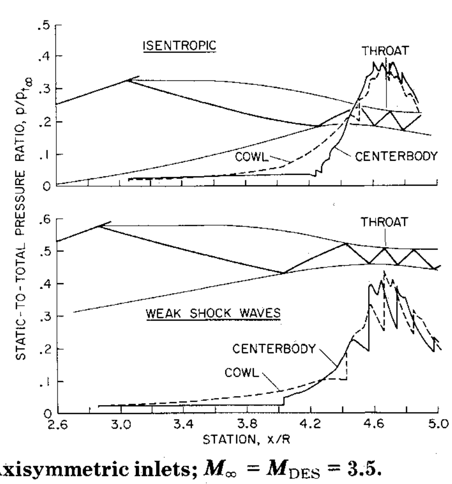

Fig. 4 Axisymmetric inlets; M_∞ = M_{DES} = 3.5.

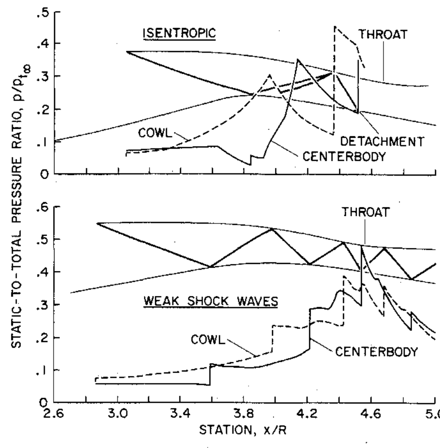

Fig. 5 Axisymmetric inlets; M_∞ = 2.75, M_{DES} = 3.5.

using the experimental data to obtain the bleed hole performance for design purposes, one must be careful that all the foregoing geometric parameters for the bleed system being designed and the experimental data base are not significantly different. The importance of this is indicated in Fig. 3 where the bleed mass-flow ratio through the hole pattern for a single centerbody plenum chamber is plotted as a function of plenum pressure ratio. The data are for a typical ⅓-scale inlet running condition (local Mach number is 1.37) while the solid curve was predicted from unpublished data. The mass-flow ratio is considerably different for different values of l/d and/or δ^*/d. For example, when the ⅓-scale inlet values of l/d and δ^*/d are, respectively, 3.3 and 2 times the predicted value, the mass-flow ratio for the ⅓-scale inlet model at any plenum pressure ratio is only half the predicted mass-flow ratio (circles). When l/d is the same but δ^*/d is twice that of the predicted values (squares) agreement is improved at least at the lower plenum pressures. Other parameters such as boundary-layer shape factor, hole slant angle and spacing, and average local surface Mach number are also important. Accurate mass flow through inlet bleed systems can be predicted only if the effect of the above indicated geometric parameters (l/d, δ^*/d, etc.) on a bleed system are known. Currently, the lack of data describing bleed hole characteristics makes it difficult to establish and verify analytical procedures, particularly at Mach numbers greater than about 3.0.

At M = 2.65, the validity of the above analytical approach has been established by large-scale inlet tests.[1,2] These same design procedures have been used to design a M = 3.5 axisymmetric inlet[8] that is scheduled for early testing in the Ames Research Center facilities. The calculated shock waves and pressure distributions for two such inlets are shown in Fig. 4 for two M = 3.5 designs. The upper inlet is designed to produce nearly isentropic compression (very weak shock waves) and the lower inlet has weak shock waves such that the theoretical inviscid throat total-pressure recovery is 0.98. The average throat Mach number for each inlet is 1.25. Note the relatively smooth pressure rise for the isentropic inlet, and the rather sharp pressure rises through the shock-wave impingements for the weak shock-wave inlet. Initially, one might choose the isentropic inlet because of its smoother static pressure distribution and slightly higher pressure recovery at M = 3.5, but at off-design Mach numbers this inlet presents problems as shown by the static pressure distributions at M = 2.75 (Fig. 5). Note the coalescence of shock waves at station 4.35 for the isentropic inlet, the associated high-pressure rises, and finally at station 4.5, lack of flow continuity (subsonic flow); the calculations stop at this point because the method of characteristics will not calculate subsonic flow and therefore the flowfield calculations are

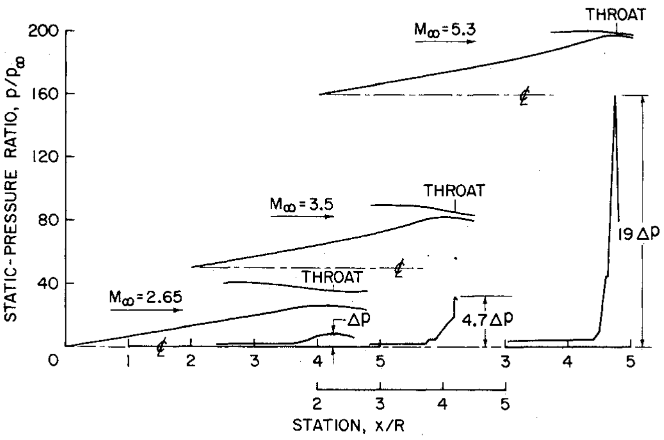

Fig. 6 Compression ratio.

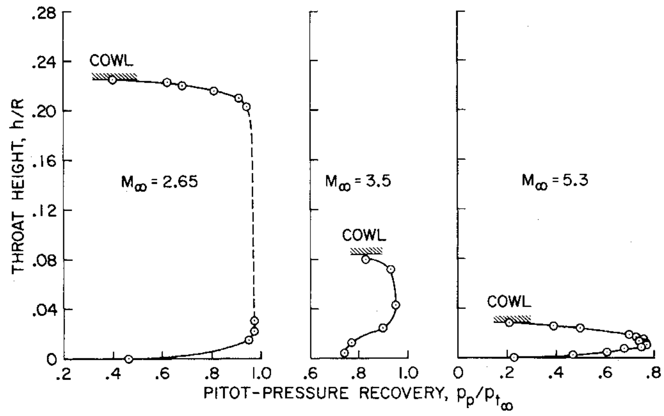

Fig. 7 Throat flow profiles; $\alpha = \beta = 0°$.

not completed to the throat. For this inlet, an efficient throat bleed system cannot be designed at this Mach number. Furthermore, efficient boundary-layer control in the supersonic diffuser will be difficult to achieve because of the large local pressure rises. For the weak shock inlet, however, the shock waves do not coalesce, subsonic flow does not occur in the diffuser, the stepwise pressure rises are relatively low, and calculations proceed through the throat—a decided improvement over the isentropic inlet. Therefore, one must be careful in choosing design contours if efficient off-design performance is an important consideration. The results suggest that high performance at off-design Mach numbers can be achieved with little sacrifice in on-design performance.

Advanced Inlet Technology Needs

The difficulty in designing bleed systems will no doubt become greater as the design Mach number increases because, if the compression is accomplished in approximately the same distance, the local rate of compression increases drastically, tending to make the avoidance of separation of the boundary layer more difficult. This increase in compression rate is illustrated in Fig. 6; the static pressure distributions on the centerbodies of three typical axisymmetric inlets designed for $M = 2.65$, 3.5, and 5.3 are shown. The static pressure rise to the throat for the $M = 3.5$ inlet is 4.7 times greater, and for the $M = 5.3$ inlet, 19 times greater than for the $M = 2.65$ inlet. Clearly, the problem of designing efficient bleed systems for advanced inlet systems will become more difficult; high bleed mass-flow rates will be required, the boundary layer will be more difficult to predict accurately, and satisfying bleed requirements for both cruise and off-design operation will be a challenging task.

Along with the need to improve the boundary-layer-control technology, the computational procedures used to define the flowfield throughout the entire region of the inlet passages must be improved. As the design Mach number increases, the throat height for a fixed capture area inlet decreases and the boundary layer encompasses a greater portion of the flow area throughout the inlet passage. This characteristic is shown in Fig. 7 where pitot-pressure profiles are plotted in the throats of the three inlets shown in Fig. 6. At $M = 2.65$, the boundary layer is only 20% of the throat height; at $M = 3.5$, it is approximately 50%; and at $M = 5.3$, the flow is nearly all boundary layer. New methods[9] that combine both inviscid and viscous flow equations must now be derived so that flowfields where the flow may be all boundary layer can be accurately predicted.

The foregoing discussion has considered only performance at 0° angle-of-attack for which the analytical procedures are valid. For inlets operating in the supersonic speed range, the flow incidence at the inlet face in flight may be as high as 5°; hence detailed flow information is not available and present procedures are not valid. At present, satisfactory performance at angle of attack is being developed through extensive wind-tunnel tests using cut-and-try techniques. However, analytical procedures for predicting performance at angle-of-attack are only now becoming available. The development of these methods is much more difficult than for $\alpha = 0°$.

Finally, many wind tunnels suitable for large-scale testing do not have the Mach number range needed for advanced supersonic inlet testing. To overcome this problem, the feasibility of locally expanding wind-tunnel flow to higher Mach numbers has been investigated. Small-scale tests have shown that a suitable flow region for testing inlets can be developed.[10] A proposed installation for the Ames 8- by 7-Foot Supersonic Wind Tunnel is shown in Fig. 8. The expansion plate has side walls to create uniform two-dimensional flow at the exit plane, and the system is rotated about the exit plane by means of an actuator. The model is mounted at the exit of the expansion plate with its center of rotation at the inlet cowl lip. With the maximum freestream Mach number of 3.5, the exit plane Mach number can be increased to 4.5 at an expansion plate angle of approximately 13°. The Reynolds number within this flowfield appears large enough to insure natural transition of the boundary layer within the inlet before inlet shock waves impinge on the boundary layer. Further, the static pressure is sufficiently low and the static temperature sufficiently high to avoid liquefaction of the air. Thus, it is possible that the maximum Mach number of the 8- by 7-Foot Wind Tunnel can be increased from the existing 3.5 to 4.5 for large-scale inlet tests.

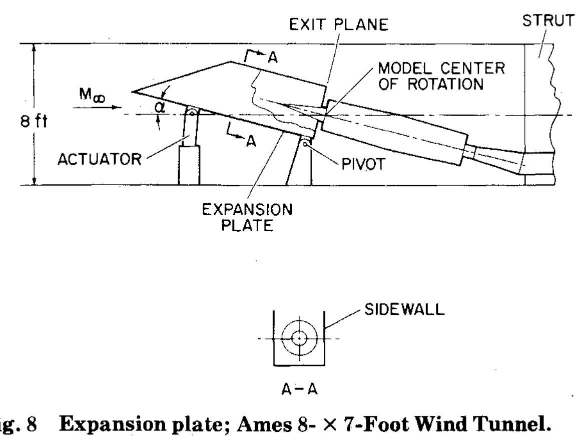

Fig. 8 Expansion plate; Ames 8- × 7-Foot Wind Tunnel.

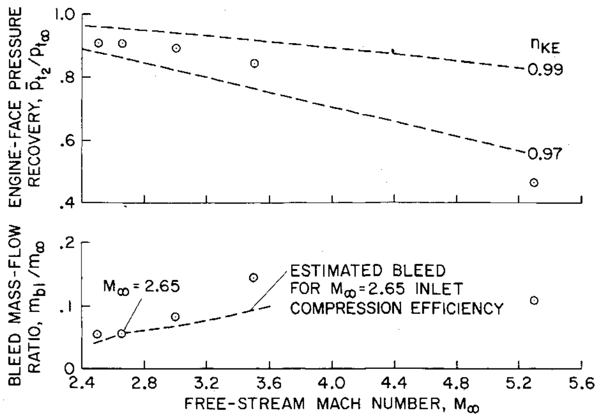

Fig. 9 Experimental cruise performance; axisymmetric inlets.

Inlet Performance

The current state-of-the-art for the performance of typical large-scale mixed-compression axisymmetric inlet systems is shown in Fig. 9. Engine-face pressure recovery and bleed mass-flow ratio at design conditions are plotted as a function of Mach number from 2.5 to 5.3 and $\alpha = \beta = 0°$. As expected, the required bleed increases with increasing Mach number for these systems up to at least $M = 3.5$.[11] The $M = 5.3$ inlet[12] has relatively low bleed but also relatively low pressure recovery.

To provide a basis for evaluating inlet performance, curves of constant kinetic energy efficiency η_{KE} are plotted in Fig. 9. With the exception of the $M = 5.3$ inlet, less than 3% of the kinetic energy efficiency of the engine-face flow is lost for any of the inlets. Note, however, that the accompanying momentum loss of the bleed flow is a major factor affecting the total inlet compression efficiency and this is not included in the η_{KE} shown.‡ Using the $M = 2.65$ inlet as a basis of comparison, the bleed allowed for equal total compression efficiency is plotted (the average pressure recovery for the bleed flow is assumed to be 25% of the engine-face recovery). Then, for each inlet, it is seen that the bleed must be reduced to attain the same compression efficiency as the $M = 2.65$ inlet. Even at $M = 2.5$,[13] the bleed mass-flow ratio should be reduced 0.015; at $M = 3.5$, the bleed should be lowered by 0.05 (calculations were not made above $M_\infty = 3.5$). To achieve these low bleed levels, very careful analytical and experimental studies will be required that embody all the previously discussed analytical procedures. It should be indicated that at Mach numbers of approximately 4 and above, the bleed required to obtain the contraction ratios necessary for reasonable pressure recovery for inlets designed for subsonic burning engines is unknown; furthermore, the bleed flow will have to be cooled. Another consideration for this Mach number range is that inlets can be designed for supersonic burning (scramjet systems); here, the lower compression ratios required may eliminate the need for boundary-layer bleed. In any event, sophisticated analytical procedures must be used to attain high performance for advanced inlet systems and, additionally, more testing will be required to verify these analytical procedures.

‡Note, too, that $(\eta_{KE})^{1/2}$ is the percentage of recoverable momentum (compression efficiency) of the engine-face flow if this flow is fully expanded to freestream static pressure through an ideal nozzle.

Conclusions

Relatively new analytical procedures have allowed prediction of efficient bleed system requirements for inlets designed for $M = 2.65$. These analytical procedures constitute a major advance in inlet technology and promise to save many wind-tunnel testing hours by avoiding much of the usual cut-and-try wind-tunnel development. However, for inlets designed to operate at Mach numbers above about 3.0, there are insufficient data to verify the new analytical procedures because bleed hole performance characteristics and detailed boundary-layer characteristics in regions of high adverse pressure gradients with bleed and shock interactions are unknown. In addition, the compatibility of cruise and off-design bleed requirements will be a more challenging problem. Moreover, new methods that couple both inviscid and viscous flow using common equations will be required to accurately predict the flowfields in the throat where there may be little or no inviscid flow. Indeed, the analytical procedures have not been verified beyond $M = 2.65$ and more testing is definitely required to substantiate the new design approach.

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