Performance Estimates for a Supersonic Axisymmetric Inlet System

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Results of recent tests of a large-scale axisymmetric mixed-compression inlet system designed for Mach number 2.65 indicate that the performance of the supersonic diffuser and required boundary-layer bleed system can be accurately estimated with relatively new analytic methods. These methods use complex computer programs to calculate both the inviscid and viscous flowfields including the effects of boundary-layer bleed. A bleed system was designed using these programs and the flow characteristics of bleed holes determined from isolated wind-tunnel tests that partially simulated the surface conditions in the inlet. These analytic methods show promise of saving many wind-tunnel testing hours by avoiding much of the usual "cut and try" wind-tunnel development.

Nomenclature

A = area
M = Mach number
m = mass flow
N = boundary-layer power-law exponent
p = static pressure
p_p = pitot pressure
p_{pl} = plenum chamber pressure
p_t = total pressure
$\bar{p}_t$ = area-weighted average total pressure
R = cowl lip radius
U = velocity
x = axial station measured from tip of the centerbody
y = distance from surface
α = angle of attack
δ = ratio of engine-face total pressure to standard sea-level static pressure
θ = ratio of freestream total temperature to standard sea-level static temperature

Subscripts

bl = bleed
c = capture
CRIT = critical
e = edge of boundary layer
L = local inviscid flow
max = maximum
min = minimum
OP = operating
th = throat
2 = engine face
∞ = freestream

Introduction

RESULTS of recent tests of a large-scale axisymmetric mixed-compression inlet system[1] have shown the value of relatively new analytical methods[2, 3] for estimating the performance of the supersonic diffuser and the required boundary-layer bleed system. If inlets can be designed with sufficient confidence to proceed directly to large-scale wind-tunnel testing, many testing hours can be saved by avoiding much of the usual "cut and try" wind-tunnel development.

The main objective of this paper is to assess the accuracy of performance estimates, mainly at the cruise Mach number, by comparisons with wind-tunnel data obtained from tests of a large-scale axisymmetric inlet model.[1]

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Model

The model is shown mounted in the wind tunnel in Fig. 1. The capture diameter was approximately 50 cm, and the centerbody translated for operation at off design Mach numbers. The centerbody is shown in the transonic position. The inlet operated in the "started" mixed compression mode down to Mach number 1.6. Below Mach number 1.6 the inlet operated in the external compression mode where the analytical methods do not apply.

More detail of the inlet can be seen in Fig. 2. The inlet was designed for shock on lip at Mach number 2.65 and isentropic compression to Mach number 1.25 in the throat. The shaded areas show where rows of boundary-layer bleed holes were located. The inlet had approximately the maximum throat area possible at transonic speeds for a system with only a translating centerbody for off-design Mach number operation. The throat station for this design remained fixed on the cowl during the started mode of operation. Because the throat remained fixed on the cowl a "travelling" centerbody bleed system was required to locate bleed opposite the cowl throat station as the centerbody was translated. The centerbody compartmentation allowed this bleed to occur as each compartment traveled over the slot in the centerbody support tube. The cowl bleed was collected in three separate chambers and then exited overboard through annular slots. The centerbody bleed was collected in two separate plenum chambers, ducted through the centerbody support tube, then through the four hollow equally spaced centerbody support struts, and finally exited overboard through the louvers at the

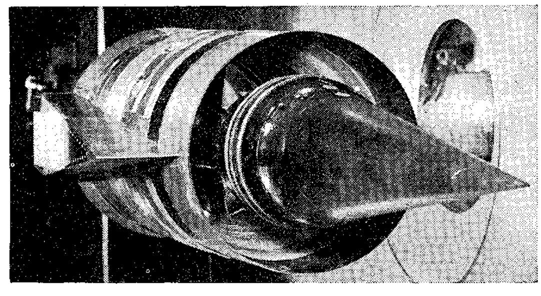

Fig. 1 Model.

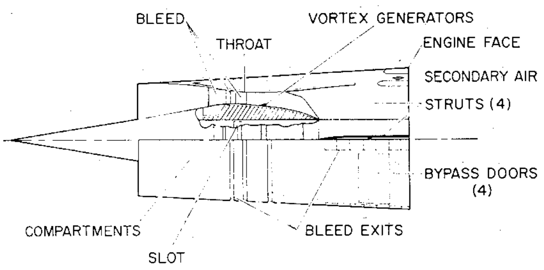

Fig. 2 Axisymmetric inlet system; $M_\infty = 2.65, \alpha = 0°$.

end of each strut. The data that will be presented are for closed bypass doors, and with a secondary airflow rate simulating minimum required engine cooling airflow. The vortex generators located just downstream of the throat on the centerbody were required to control flow distortion at the engine face.

The plenum chambers were compartmented to prevent the high pressure bleed airflow downstream from recirculating into the low pressure upstream regions and causing separation of the boundary layer. This also allowed higher pressure recoveries in each plenum chamber than would be possible without compartmentation thereby minimizing the momentum drag of the overboard bleed flow. By slanting the bleed holes (Fig. 3) the bleed pressure recovery can be increased over that for holes normal to the surface. Ideally, all the holes should be slanted 20° or less to maximize the plenum chamber pressure recovery, but the 20°, 40°, and 90° hole angles, shown in the figure, were used in this inlet as a compromise for the operating condition, as will be mentioned later. All holes were 0.159 cm diam, which was approximatly half of the local boundary-layer height. The ratio of hole size to boundary-layer height was believed to be small enough for the required performance while facilitating filling or reopening rows of holes for various changes in the bleed pattern during testing. Eight pitot-pressure rakes were located just ahead of and behind each bleed region to measure the boundary-layer profiles.

Comparisons of Estimates with Experiment

Contours

One of the first steps in any inlet system design is the determination of the supersonic diffuser contours. The present contours were designed with the aid of an inviscid computer program[4] using only the supersonic method of characteristics and did not account for the boundary layer. Results of previous tests on other similar inlets[5] indicated that the usual amounts of bleed flow required, to prevent boundary-layer separation, closely compensated for the blockage effects of the boundary layer. This method of design is validated by

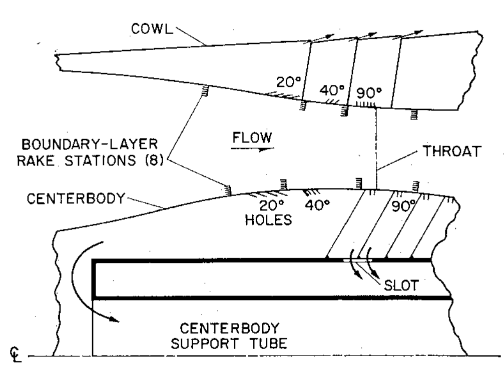

Fig. 3 Bleed system; $M_\infty = 2.65, \alpha = 0°$.

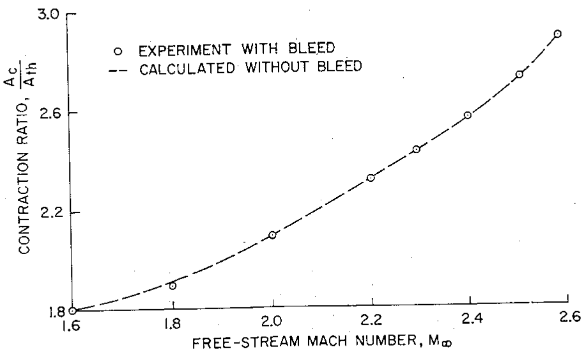

Fig. 4 Critical contraction ratio.

comparisons in Fig. 4 of the calculated and experimental contraction ratios occurring just before the inlet throat area chokes, and unstarts the inlet at Mach number 1.6 to 2.60. The terminal shock wave was well downstream of the throat for the experimental data and did not affect the unstarting of the inlet. Agreement is very good except at Mach number 1.8 where surface and bleed plenum chamber pressures indicated possible recirculation of the flow, resulting in excessive boundary-layer blockage. This blockage was not completely compensated for by bleed, causing the inlet to unstart at a slightly lower contraction ratio than calculated.

Pressure Distributions

How well the boundary layer was controlled can be shown by how closely the experimental pressure distributions agree with the distributions calculated by the inviscid computer program. A typical example is shown in Fig. 5 where the experimental and calculated pressure distributions are plotted for the cowl at Mach number 2.65 and 0° angle of attack. Experimental distributions are shown with the terminal shock wave far downstream in the supercritical position, at an operating position with the terminal shock wave in the throat region, and at the critical position just before the inlet unstarts. Agreement is good ahead of the bleed areas where pressure gradients are not high enough to separate the boundary layer. The fair agreement in the bleed regions is probably due to local separation caused by the high pressure gradients even with the boundary-layer bleed or the local disturbances caused by turning of the flow into the bleed holes or both. Even with these discrepancies the boundary layer was controlled well enough to prevent separation in the throat with the terminal shock wave in an operating position as evidenced by the sharp pressure rise at $x/R = 4.20$. At the

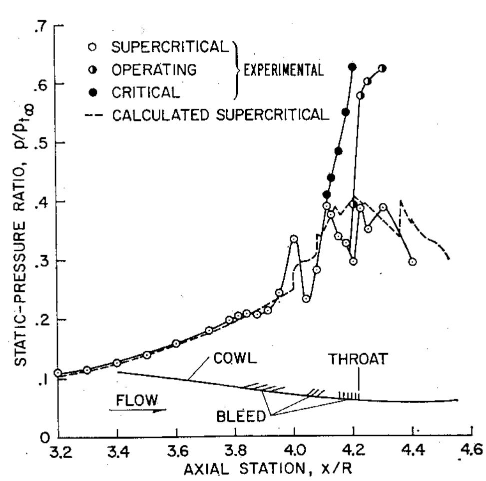

Fig. 5 Cowl pressure distribution; $M_\infty = 2.65, \alpha = 0°$.

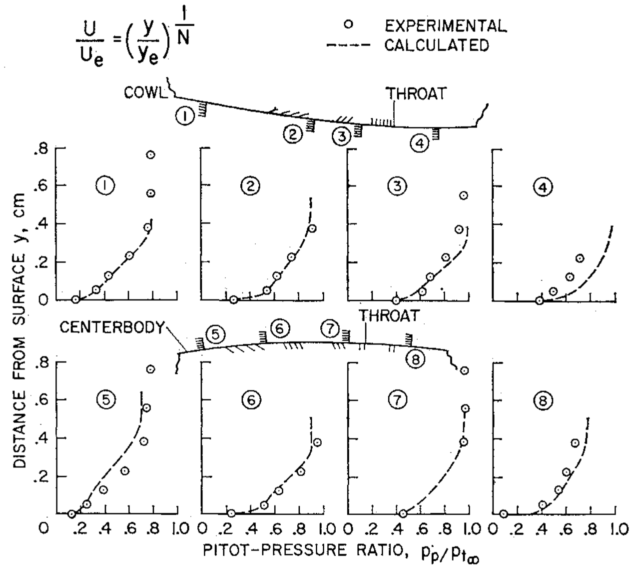

Fig. 6 **Boundary-layer profiles;** $M_\infty = 2.65, \alpha = 0°$.

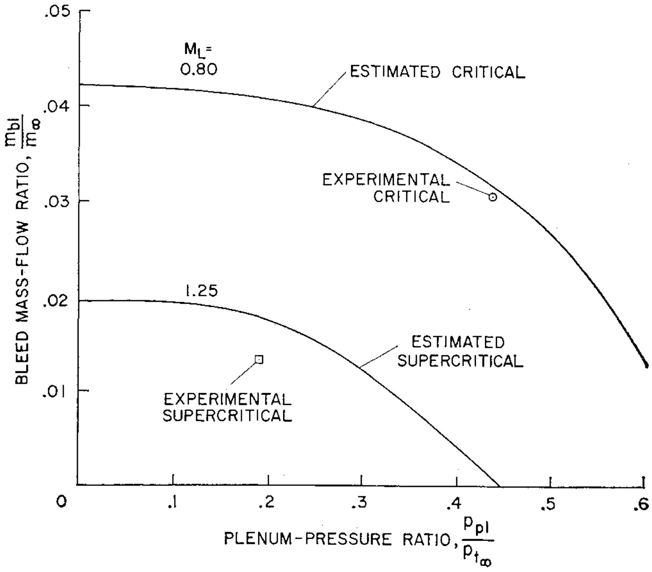

Fig. 7 Cowl throat bleed; $M_\infty = 2.65, \alpha = 0°$.

critical position the boundary layer was not as well controlled in the throat as evidenced by the more gradual pressure rise starting at $x/R = 4.11$.

Boundary Layers

One of the most important aspects in the analytical development of an efficient bleed system is accurate prediction of the boundary-layer characteristics. Based on these characteristics, bleed is incorporated in the regions of probable boundary-layer separation. The criterion for boundary-layer separation in the present analysis[3] is based on maintaining limits on the value of the exponent N in the familiar boundary layer velocity profile equation shown in Fig. 6. Experience has shown that to prevent boundary-layer separation, N should be approximately 3 or greater in the supersonic diffuser and approximately 7 or greater in the throat. The boundary layer velocity profile is first calculated without bleed.[2] Then bleed is incorporated so that the exponent for the profiles is within the acceptable range to prevent separation. This procedure could require several iterations. The final predicted boundary-layer pitot-pressure profiles are compared with experimental profiles in Fig. 6 at Mach number 2.65° and 0° angle of attack at eight stations along the cowl and centerbody. The agreement is considered quite good with no experimental evidence of boundary-layer separation. This might be expected for those profiles upstream of any bleed. For the remaining profiles, downstream of the bleed regions, agreement is considered accurate enough to confidently design an adequate bleed system.

Bleed Holes

Another important aspect of the analytical method is accurate estimation of the boundary-layer bleed flow rate and the maximum plenum chamber pressure recovery at which the flow can be discharged overboard. If the estimates are not accurate, the bleed flow rate will be either inadequate or unnecessarily high. An example of the accuracy is shown in Fig. 7. Estimated† and experimental bleed mass-flow ratios are plotted as a function of plenum chamber pressure for the 90° holes on the cowl. Results are shown at the critical condition just before the inlet unstarts and at the supercritical condition with the terminal shock wave well downstream of the throat. The estimates are based on experimental data for the flow characteristics of similar holes for each local inviscid flow Mach number. At critical conditions, the experimental result is close to the estimated curve. At the supercritical condition, however, the experimental result is less than estimated because the experimental local inviscid flow Mach number was greater than the calculated Mach number of 1.25.

As a further comparison of the accuracy of the bleed estimates, Table 1 compares the experimental and estimated bleed flow rate and plenum chamber pressure at the critical condition for each bleed plenum on both the cowl and centerbody. The total experimental cowl bleed mass-flow ratio was 0.006 less than estimated (0.055 vs 0.049) and the total centerbody bleed 0.002 less (0.033 vs 0.031). Note that the differences for each plenum chamber are not compensating, with the estimated values of the bleed all slightly higher or equal to the experimental values. Differences in the plenum chamber pressure appear to be random. All estimates, however, are considered sufficiently accurate to allow good estimation of the bleed drag.

Performance Summary

Other bleed hole configurations,[1] only slightly different from the predicted configuration, were tested in an effort to improve the performance. The experimental results for the model with these changes incorporated are shown in Table 2. Performance for an initial bleed configuration (18) and two of the other better configurations (22 and 20) are summarized

Table 1 Critical bleed flow and recovery; $M_\infty = 2.65, \alpha = 0°$

Cowl plenum	m_{bl}/m_∞		$p_{tbl}/p_{t\infty}$	
	Est	Exp	Est	Exp
Forward	0.014	0.011	0.185	0.204
Mid	0.010	0.009	0.195	0.175
Throat	0.031	0.029	0.450	0.420
$\Sigma m_{bl}/m_\infty$	0.055	0.049		
Centerbody				
Forward	0.016	0.014	0.140	0.166
Throat	0.017	0.017	0.450	0.479
$\Sigma m_{bl}/m_\infty$	0.033	0.031		

† The estimated bleed mass-flow ratios were obtained from isolated tests that partially simulated the hole configurations and boundary-layer profiles in the inlet. The results of the tests (shown in Fig. 7 as estimated) are only partially published here and in Refs. 1 and 3.

Table 2 Experimental performance; $M_\infty = 2.65$, $\alpha = 0°$

Configuration	Critical			Operating		
	Recovery, $\bar{p}_{t2}/p_{t\infty}$	Total bleed, m_{bl}/m_∞	Distortion, $\frac{(p_{t2})_{max} - (p_{t2})_{min}}{\bar{p}_{t2}}$	Recovery, $\bar{p}_{t2}/p_{t\infty}$	Total bleed, m_{bl}/m_∞	Distortion $\frac{(p_{t2max} - (p_{t2}))_{min}}{\bar{p}_{t2}}$
18	0.937	0.081	0.093	0.905	0.064	0.143
22	0.941	0.089	0.074	0.910	0.071	0.131
20	0.943	0.094	0.070	0.913	0.077	0.124

[a] Control margin, $\frac{[m_2(\theta)^{\frac{1}{2}}/\delta_2]_{OP} - [m_2(\theta)^{\frac{1}{2}}/\delta_2]_{CRIT}}{[m_2(\theta)^{\frac{1}{2}}/\delta_2]_{OP}} = 0.05.$

for the critical and operating condition. The operating condition is defined as the point where a 5% step decrease in engine corrected airflow can occur without unstarting the inlet.‡ The changes made in the wind tunnel did raise the engine-face pressure recovery slightly but with an unfavorable increase in bleed flow. (It is estimated that 0.01–0.02 increase in pressure recovery is required to offset 0.01 increase in bleed mass-flow ratio.)[3] At the operating condition the trade of bleed for recovery is less favorable for configurations 22 and 20 than for the initial configuration 18. Also at the operating condition the distortion for all configurations may be unacceptably high, but it is known that the distortion can be reduced by the addition of vortex generators on the cowl.[1] However, the initial configuration designed with the aid of the analytical methods appears to give good over-all performance.

Concluding Remarks

The results indicate that relatively new analytical methods show promise for reliably estimating the performance of the supersonic diffuser of mixed-compression axisymmetric inlet systems up to about Mach number 3.0. The contours and the performance of the required boundary-layer bleed system have been accurately predicted for at least one inlet system so that maximum pressure recovery with a minimum of bleed mass flow and overboard bleed drag can be achieved quickly in the wind tunnel with only minor changes to the bleed system. Thus, many testing hours can be saved by avoiding much of the usual "cut and try" wind-tunnel development.

‡ To allow this step decrease, 90° holes are required in the throat because the change in bleed flow for 90° holes is as much as three times greater than for slanted holes as the terminal shock wave moves upstream over the holes.

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