AN EXPERIMENTAL STUDY OF PERFORATED INTAKE DIFFUSERS AT A MACH NUMBER OF 2.50

by

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SUMMARY

A two-dimensional and an axisymmetric diffuser of the reversed de Laval nozzle type with perforations on the convergent portion were tested in the UTIA 16 in. x 16 in. supersonic wind tunnel at a free-stream Mach number of 2.50 and a Reynolds number of 2.50 x 10^5 per inch in order to study their operating characteristics.

The current description of the shock swallowing process of such diffusers is based on the simplifying assumption of a single normal shock. Schlieren photographs have shown that in actual fact a single normal shock is non-existent. What was observed was a prominent lambda shock in the convergent portion and multiple shocks in the divergent portion. It was shown through high-speed motion pictures that these shock waves and the flow in the diffuser were oscillatory.

In tests of the two-dimensional diffuser, when the lambda shock was swallowed, no total pressure losses were detected between its inlet and the multiple shocks in its divergent portion. It was found that the static pressures measured along the diffuser axis and on the wall coincided with the pressure distribution of an isentropic flow in a reversed de Laval nozzle except in the vicinity of the throat and beyond where the overall total pressure losses across the multiple shocks, however, were much higher than those expected from the normal shock theory.

In testing the axisymmetric diffuser, it was found that a long throat did not reduce the total pressure losses, and that the most effective location of the perforations was in a region just upstream of the throat.

Perforated intake diffusers are to date not very useful in practice because of the large spillage in their convergent portion and the high total pressure losses in their divergent portion. Some improvement may be achieved by paying more attention to the design of the diffuser contour. In the convergent portion, the perforations should be distributed such that the shock swallowing process is accomplished with minimum spillage, and in the divergent portion the losses due to flow separation should be avoided. These problems, although recognized in the present work, were not pursued further.
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SYMBOLS

m  Mass flow (slugs/sec.)

m_0  Mass flow through the entrance under free-stream conditions (slugs/sec.)

T  Total temperature (deg. R.)

R  Gas constant (1716 ft.lb./slug, deg. R.)

A  Area (sq. ft.)

\gamma  Ratio of specific heats (\gamma = 1.4)

P  Total or isentropic stagnation pressure (lbs./sq.ft.)

p  static pressure (lbs./sq.ft.)

M  Mach number

F(M) = \frac{m\sqrt{T}}{A_P} \sqrt{\frac{\gamma}{\gamma - 1}}  Fliegner number (a function of Mach number)

X = M_m / M_o  Mass flow recovery

Y = P_m / P_o  Total pressure recovery

Q_a  Subsonic flow coefficient (ratio of the effective to perforated area downstream of a normal shock)

Q_b  Supersonic flow coefficient (ratio of the effective to perforated area upstream of a normal shock)

\omega  Prandtl-Meyer angle (deg.)

\alpha  Mach angle (deg.)

A_p / A_t  Perforation ratio (perforated area/throat area)

A_e / A_t  Exit ratio (plug exit area/throat area)

Subscripts

o  Free-stream

m  Measuring plane
(iii)

* Reversed de Laval nozzle

e Exit

th Throat

p Perforation

x Position x along longitudinal axis of diffuser
I. INTRODUCTION

The performance of a ramjet engine is primarily determined by the efficiency of its intake diffuser. The diffuser efficiency depends on the total pressure recovered from the free-stream total pressure and on the mass flow passing through the diffuser exit. Any loss in total pressure results in a poorer thermal efficiency. Any spillage of mass flow at the inlet or through perforations in the wall of the diffuser is responsible for a loss of internal thrust and also for an increase of external drag. Present supersonic intake diffusers may be divided into four groups: normal shock diffusers, spike diffusers, convergent-divergent diffusers and perforated diffusers. Perforated diffusers offer some characteristics that are superior to those of the others. For example, they achieve total pressure recoveries higher than normal shock diffusers, they are mechanically simpler than spike diffusers and they reduce the complexity of swallowing the shock at isentropic contraction ratios. As these contraction ratios are higher than those for convergent-divergent diffusers, the throat Mach number is smaller and higher total pressure recoveries are obtained. Nevertheless, the perforated diffuser has not yet achieved the status of a useful intake in practice. Experimental evidence (Refs. 4 and 5) at higher Mach numbers (2.50) showed that the actual mass flow and total pressure recovery values fell short of those predicted theoretically (Ref. 1). A further study of the perforated diffuser was therefore considered necessary in an effort to gain a better understanding of its basic operation in order to obtain a more realistic theoretical model and some guidance to possible practical improvements.

Perforated diffusers have been tested at the NACA and UTIA. In 1947, Evvard and Blakey investigated a perforated diffuser at a free-stream Mach number of 1.85 (Ref. 1). This diffuser had a geometric internal contraction ratio of 1.49, which is the isentropic value. The convergent wall of the diffuser was of arbitrary shape but gently curved and the divergent portion had an included angle of 5 degrees. Many countersunk holes were drilled at random through the wall of the convergent portion. It was shown experimentally that a perforated reversed de Laval nozzle can eliminate the difficulty in swallowing a normal shock and consequently allow for higher total pressure recoveries.

In addition to their tests, Evvard and Blakey formulated a theory for the distribution of perforations along the convergent portion of a diffuser at any free-stream Mach number. This theory was based on the assumptions that (1) the flow is one-dimensional in the inlet, (2) the local Mach number is the same as that in a reversed de Laval nozzle, (3) the total pressure loss in the inlet is due to a normal shock, and (4) the shock is temporarily stable in the convergent portion of the diffuser. Downstream of the shock, the pressure ratio across the holes is supercritical and consequently the flow through the holes is choked. Upstream of the shock, this
pressure ratio is below critical and only a small amount of supersonic flow can be spilled out through the holes by way of a Prandtl-Meyer fan. Thus, the location of the shock in the convergent portion controls the total amount of spillage through the perforations. By considering the location and the strength of the shock, the theory shows that if the perforations are distributed accordingly, the shock can be positioned at the throat of the diffuser. As a result, a higher total pressure recovery can be obtained at a minimum spillage through the perforations.

Hunczak and Kremzier continued the above investigation at NACA in 1950 (Ref. 2). Eight perforated diffusers having various contraction ratios were tested at a free-stream Mach number of 1.90. Internal contours of all inlets were designed by choosing an almost linear reduction in local Mach number from the entrance to the throat of the diffuser. The experimental values of maximum mass flow recovery were in substantial agreement with those predicted by the Evvard-Blakey theory.

Weinstein enlarged the scope of those investigations by looking into the effect of a boundary layer developed in a cylindrical pipe which was connected to the upstream end of the inlet of perforated diffusers (Ref. 3). These diffusers were of the same geometrical configuration and were tested under the same conditions as those of Ref. 2. It was found that on account of this initial boundary layer, the maximum mass flow recovery was decreased by approximately 2 percent. In addition to the above tests, a supplementary experiment was carried out in which a straight cylindrical pipe was coupled to the entrance of the divergent part (throat) of the diffuser. It appeared that the total pressure recovery varied with shock position in this pipe. When the shock was located close to the pipe entrance, the total pressure recovery was the same as that across a normal shock at the free-stream Mach number. With the shock positioned close to the pipe exit, the total pressure recovery was much lower. It was suspected that the interaction of shock waves with the boundary layer induced flow separation in the divergent portion which is responsible for the reduced total pressure recovery.

At UTIA, an experimental investigation of the total pressure recovery of an axisymmetric perforated diffuser at a free-stream Mach number of 2.50 was conducted by Brown in 1957 (Ref. 4). The contour of the convergent portion of the diffuser was designed by Foelsch's method (Ref. 6), and the divergent portion had an included angle of 7 degrees. Rows of holes were drilled starting close to the entrance and proceeding toward to the throat until the peak total pressure recovery was obtained. The final ratio of the perforated area to the throat area was 4.28. The maximum mass flow recovery determined experimentally was 88% as compared with the theoretical estimate of 77% from the Evvard-Blakey theory.

Reference 5 describes a further UTIA investigation of a two-dimensional perforated diffuser at a free-stream Mach Number of 2.50 by Clark in 1958. The internal contour of the convergent portion of this diffuser
was based on the design of a supersonic de Laval nozzle (Ref. 7), and the divergent part had an included angle of 7 degrees. This model had two windows installed on the side walls of the diffuser to facilitate schlieren studies of the shock swallowing process. Many rows of small holes were drilled between the inlet and the throat of the diffuser and these holes were subsequently enlarged until the perforated area was large enough for the shock to be apparently swallowed as determined from schlieren photographs. However, a further investigation of the same model by the present author has in fact shown that the shock was still not completely swallowed. This was found by using a schlieren system with a pulsed-light source of short duration.

From the review of previous work, it can be seen that the working principle of the perforated diffuser with an isentropic contraction ratio is to force the 'normal shock' into a position downstream of the throat by the use of perforations. When the shock is downstream of but close to the throat, the peak total pressure recovery and the maximum mass flow recovery are obtained. The theoretical distribution of perforations proved to be correct for the NACA models at a free-stream Mach number of 1.90 but there was disagreement with the UTIA perforated diffusers at a free-stream Mach number of 2.50. It seems that the Evvard-Blakey theory is valid at low Mach numbers but that at higher Mach numbers, it appears that boundary layer effects can no longer be neglected on the shock swallowing process. As to the design of the internal contour of perforated diffusers, NACA models were of an arbitrary shape with an almost linear reduction in local Mach number and UTIA models were designed as reversed de Laval nozzles for uniform flow. Further information for the design of diffuser contours is required for improving the performance of perforated diffusers.

The content of the present study is as follows: (1) to investigate the generated shock patterns and in particular to find whether they are normal and stationary under the influence of the boundary layer, (2) to examine the validity of the assumption of one-dimensional flow in the diffuser, (3) to analyse the cause of the total pressure loss along the diffuser axis, (4) to try to increase the peak total pressure recovery by the use of a long throat and (5) to determine the effectiveness of perforations as a function of position between inlet lip and throat.

II. THEORETICAL ANALYSIS

2.1 Total Pressure Recovery and Mass Flow Recovery

2.1.1 Fliegner Number

If the flow in a perforated diffuser is one-dimensional, the flow properties at any section are related by the following equation as derived in Ref. 8:
The left-hand side of this equation is a non-dimensional parameter including such fluid properties as the mass flow (m), the total pressure (P) and the total temperature (T) which are specified at a sectional area (A). This parameter, first used empirically by Fliegner (Ref. 9) and tabulated as a Mach number function in Ref. 10, is quite helpful when discussing perforated diffusers and will subsequently be referred to as the Fliegner number F or F(M). The graph of Fliegner number (F) vs. Mach number (M) is plotted in Fig. 3. Note that the Fliegner number reaches its maximum (0.5787) when the Mach number is unity.

2.1.2 Total Pressure Recovery and Mass Flow Recovery at the Measuring Plane

The total pressure recovery (Y) is defined as the ratio of the total pressure (P_m) measured at a measuring plane (Figs. 1 and 2) to the free-stream total pressure (P_o). The mass flow recovery (X) is defined as the ratio of the mass flow (m_m) passing through the throat of the diffuser to the mass flow (m_o) through the inlet area under free-stream conditions. The total pressure recovery and the mass flow recovery at a measuring plane of a perforated diffuser can be written in combination with Eq. 1 as,

\[
\frac{m\sqrt{T} \sqrt{R}}{A \sqrt{P} \sqrt{\frac{T}{T}}} = \sqrt{\frac{2}{T-1}} \left( \frac{P}{P_o} \right) \left( 1 - \frac{P}{P_o} \right)^{\frac{T-1}{2}} = M \left[ 1 + \frac{Y-1}{2} \right]^{\frac{1}{2}} = F(M) \tag{1}
\]

From Eqs. 1 and 4, it can be seen that the ratio of mass flow recovery and total pressure recovery at the measuring plane is a function of Fliegner number. If the flow at the measuring plane is one-dimensional, Eq. 4 provides a simple method for determining the mass flow.
flow recovery from the total and static pressures measured at any point in the measuring plane. Since, however, the flow at the measuring plane is not exactly one-dimensional, the average total pressure and Mach number have to be used in Eq. 4 to give the mass flow recovery.

2.1.3 Total Pressure Recovery and Mass Flow Recovery at the Exit Plug

As shown in Figs. 1 and 2 and Plate 1, the perforated diffuser models have a plug after the simulated combustion chamber to vary the mass flow passing through it. The Mach number at the plug exit may be assumed to be unity because the pressure ratio across the plug is always well above the critical one. The effective area at the plug may be considered as approximately equal to the geometrical area since the boundary layer developed on the plug and wall is probably small owing to the favorable pressure gradient. Hence, the following equation (from Eq. 2) applies at the plug exit,

\[ \frac{X}{Y} \frac{m_0 \sqrt{T_0}}{A_e P_0} \sqrt{\frac{R}{\gamma}} = F(1) = 0.5787 \tag{5} \]

where the value of 0.5787 may be obtained from Eq. 1 or from Fig. 3. Similarly, referring to the conditions at the throat of a reversed de Laval nozzle, one gets,

\[ \frac{m_0 \sqrt{T_0}}{A_t P_0} \sqrt{\frac{R}{\gamma}} = F(1) \tag{6} \]

Therefore, the ratio of the mass flow recovery (\(X\)) to the total pressure recovery (\(Y\)) is equal to the exit ratio (plug exit area/throat area):

\[ \frac{X}{Y} = \frac{A_e}{A_t} \tag{7} \]

This is another approximate and simple method of finding the mass flow recovery from a given exit ratio and total pressure recovery. The latter can be obtained from a pitot tube rake at the exit of the divergent portion of the perforated diffuser.

2.1.4 Total Pressure Recovery and Mass Flow Recovery at the Throat

If the measuring plane is assumed to be at the throat, the total pressure recovery and the mass flow recovery at the throat of a perforated diffuser can be related similarly to Eq. 4 as,

\[ \frac{X}{Y} = \frac{F(M_t)}{F(1)} \tag{8} \]
As long as the shock is downstream of the throat, the throat Mach number remains constant and the mass flow recovery is at its maximum. If total pressure losses occur through the shock only, \( Y \) at the throat is equal to unity. Then, the maximum mass flow recovery becomes,

\[
X_{\text{max}} = \frac{F(M_t)}{F(1)}
\]

Since the maximum mass flow recovery for perforated diffusers (because of spillage loss) is always less than one, \( F(M_t) \) must be smaller than its maximum value (see Fig. 3) and therefore the throat Mach number must be greater than unity.

Equation 9 is useful to find the theoretical values from the experimental results and vice versa. For example, if \( X_{\text{max}} \) is experimentally determined, the theoretical throat Mach number follows from Fig. 3 or from Eq. 9. Then, the theoretical peak total pressure recovery can be calculated from this throat Mach number by assuming that the normal shock occurs at the throat. This means that the theoretical peak total pressure is numerically equal to the total pressure ratio across a normal shock at the theoretical throat Mach number and that it is the maximum possible total pressure recovery for the diffuser. On the other hand, if the throat Mach number is determined from wall static pressure measurements as described in Sec. 4.1.3, the theoretical maximum mass flow and maximum possible total pressure recovery can also be found.

2.2 Subsonic and Supersonic Flow Coefficients

In Ref. 1, the subsonic and supersonic flow coefficients (\( Q_a \) and \( Q_b \) respectively) are defined as the ratios of the effective area to the actual perforated area downstream and upstream, respectively, of a normal shock in the convergent portion of a diffuser. In Ref. 1, \( Q_a \) was assumed to be an arbitrary constant. However, it was found that \( Q_a \) can be determined experimentally as follows: If a bow shock is detached from the lip of a perforated diffuser, regardless of the size of the exit plug opening, the diffuser is said to be under-perforated. Under this condition, the flow spilled out through the perforations is sonic because the pressure ratio across the perforations is well above the critical value. If, with the exit plug area at its maximum, the total perforated area is increased to what one might call the minimum perforated area, the shock will be just attached to the lip. In this case, the flow through the throat and through the perforations has a Mach number of unity and the following equation (derived from Eq. 1) applies,

\[
\frac{m \sqrt{T_0}}{(A_0 + A_p Q_{0})} \frac{R}{c} \sqrt{\frac{R}{c}} \cdot \frac{R}{c} = F(1)
\]
where \( Y_0 \) (the total pressure recovery) is equal to the total pressure ratio across a normal shock at the free-stream Mach number. By combining Eqs. 6 and 10, \( Q_a \) can be determined from,

\[
Q_a = \frac{1 - \frac{Y_0}{Y_0 (\frac{A_p}{A_t})_{\text{min}}}}{(11)}
\]

where the minimum perforation ratio \( (\frac{A_p}{A_t})_{\text{min}} \) can be found with the aid of a schlieren system.

The choked flow through the perforations is similar to a jet discharged from a pipe through an orifice. Therefore, the subsonic flow coefficient \( Q_a \) is of the same nature and order as a jet contraction or jet discharge coefficient. In Ref. 9, the coefficient of contraction for a Borda orifice is given by

\[
\frac{1}{\delta M^2} ((1 + \frac{\delta M^2}{2}) \frac{\delta M}{1} - 1)
\]

which is equal to 0.64 at \( M = 1 \). As to the shape of the perforation hole, a sharp-edged orifice type, which is countersunk from the outside of the diffuser, is more effective than a straight hole of the same area (Ref. 11).

The supersonic flow coefficient \( Q_b \) was derived in Ref. 1 by assuming that the static and total pressures at the perforation exit are the same as those of the free-stream and that the thickness of the diffuser wall is negligibly thin. The expression for the calculation of the supersonic flow coefficient given in Ref. 1 can be simplified by applying the Fliegner number \( F(M) \) and Prandtl-Meyer angle \( \omega \) (see Appendix) as follows:

\[
Q_b = \frac{F(M_x)}{F(M_0)} \sin(\omega_0 - \omega_x)
\]

where \( M_x \) is a local Mach number at a position \( x \) along the longitudinal axis of the diffuser in which the flow is assumed to be one-dimensional. Based on the above assumptions, it appears that the final Mach line of the Prandtl-Meyer expansion fan, originating at a perforation may turn past the position parallel to the diffuser wall (see Fig. 5K. 1 in Appendix). In this case, the flow at the perforation exit could have a velocity component opposite to the free-stream direction. This, however, cannot happen (Ref. 2) and therefore the optimum angle of the final Mach line of the Prandtl-Meyer expansion is reached when this Mach line becomes parallel to the diffuser wall. Then, the supersonic flow coefficient with limited expansion becomes (see Appendix),

\[
Q_b = \frac{F(M_x)}{F(M_p)} \sin \alpha_p
\]

where \( M_p \), the Mach number at the perforation exit, cannot be equal to the free-stream Mach number. For example, if the free-stream Mach
number \( M_0 = 2.50 \) and the local Mach number \( M_\lambda = 1.10 \), \( M_p \) is found to be 2.12 (see Appendix) instead of 2.50.

In Fig. 4, the supersonic flow coefficient \( Q_b \) is plotted versus the local Mach number \( (M_\lambda) \) at a free-stream Mach number of 2.50. When the local Mach number drops below 1.62, the limiting condition becomes effective. Equation 12 therefore applies for local Mach numbers larger than 1.62 and Eq. 13 for those less than 1.62. It can be seen in Fig. 4 that the supersonic flow coefficient increases as the local Mach number decreases and that at local Mach numbers smaller than 1.62 supersonic flow coefficients for limited expansion are higher than those for unlimited expansion.

The supersonic flow coefficient is useful in determining the theoretical maximum mass flow recovery. The spillage through a row of perforation at any section can be written as (from Eq. 1),

\[
\frac{\Delta m \sqrt{T_0}}{\Delta A_p Q_b P_o \sqrt{\gamma}} = F(M_\lambda) \tag{14}
\]

where \( \Delta A_p \) is the increment in perforated area along the diffuser. Then, the maximum mass flow recovery can be obtained, by combining Eqs. 6 and 14, from

\[
X_{max} = 1 - \sum \frac{F(M_\lambda)}{F(1)} \left( \frac{\Delta A_p}{A_t} \right) Q_b \tag{15}
\]

2.3 Perforation Distribution

The theoretical distribution of perforations derived in Ref. 1 is based on the assumption that a normal shock exists in the convergent portion of a perforated diffuser. In practice, due to the action of the boundary layer, the normal shock is in fact a lambda shock. Consequently, this distribution of perforations is no longer valid. The flow passing through the lambda portion of the Mach reflection is supersonic along the diffuser wall. Therefore, the spillage downstream of the lambda shock is governed by the supersonic flow coefficient rather than by the subsonic one. Since supersonic flow coefficients are smaller than subsonic flow coefficients (see Sec. 2.2 and Fig. 4), more perforations are required downstream of the lambda shock than for a normal shock. This means that perforations close to the throat are more effective in their purpose of swallowing the shock. Hence, the perforations should be concentrated in the area upstream of but close to the throat of the diffuser.
To keep the spillage to a minimum, the total perforated area must be as small as possible but just large enough to complete the shock swallowing process. The determination of the optimum total perforated area is difficult because the spillage depends on the local flow properties (density and velocity) and on the effective perforated area (= geometrical area times supersonic flow coefficient). The supersonic flow coefficient, however, is a function of Fliegner number (see Eqs. 12 and 13) which in turn depends on the respective total and static pressures (Eq. 1). All these fluid properties (density, velocity and pressures) cannot be determined analytically. However, the total perforated area required is probably only a few percent larger than the minimum perforated area if the perforations are distributed in a region close to the throat. This small area increment seems to be needed to force the shock from the lip down to the throat. No attempt was made to determine theoretically the total perforated area required, but experimentally it was found (see Sec. 4.2.2) that the total perforated area required was almost equal to the minimum perforated area.

2.4 Diffuser Contours

The convergent portion of the diffuser should be designed such that the lip angle at the inlet is small and the static pressure gradient is almost constant along the diffuser. A large lip angle would cause total pressure losses due to strong oblique shocks and their reflections from the diffuser walls. A high pressure gradient upstream of but close to the throat in a rapidly converging portion provides only a narrow region to accommodate sufficient, effective perforations. As an illustration for the aforesaid, let us consider four possible axisymmetric diffuser designs; diffuser A, designed by Foelsch's method; diffuser B, a straight cone; diffuser C, with a linear reduction of Mach number from the inlet to the throat; and diffuser D, with a constant pressure gradient along the diffuser axis. Figures 5 and 6 show the diffuser contours and their static pressure distributions, respectively, on the basis of equal isentropic contraction ratios and equal diffuser lengths at a free-stream Mach number of 2.50. Diffuser A provides extreme high pressure gradients close to the throat and a zero lip angle, as the surface at the lip is in line with the free streamlines. The conical intake, B, is the simplest in shape; its static pressure rise in the region close to the throat is not as steep as that of diffuser A and its lip angle is small but depends on the length of the convergent portion of the diffuser. Diffuser C has a somewhat larger lip angle and the pressure gradient in the region close to the throat is still fairly large. Finally, diffuser D has the largest lip angle and the smallest pressure gradient. None of these four types of diffusers is good enough to satisfy the design conditions for perforated diffusers as mentioned above. The diffuser models tested at NACA were similar to diffuser C and these tested at UTIA were similar to diffuser A. What is needed is a diffuser with the shock free inlet flow of diffuser A and with a low pressure gradient close to the throat. This can be accomplished by using the method of characteristics in combination with an elongation of the convergent portion of the diffuser.
2.5 Shock Swallowing Process

The idealized model of the shock swallowing process in a perforated diffuser with an isentropic contraction ratio is based on a single normal shock through which the total pressure loss occurs. This single normal shock, however, is not verified by experimental evidence. The actual operation of perforated diffusers can be divided - similar to that of a spike diffuser (Ref. 12) - into three regimes: subcritical, critical and supercritical. The subcritical operation is characterized by a bow shock being either detached from or attached to the lip of the diffuser. In this regime, the total pressure recovery remains the same as that across a normal shock at the free-stream Mach number, and the mass flow recovery varies with the size of the exit plug opening (see e.g., Fig. 7). When the shock is located at the lip, the subsonic flow in the convergent portion is accelerated toward the throat and ideally reaches a Mach number equal to unity. In this case the mass flow recovery is numerically equal to the total pressure recovery (see Eq. 8 and point B in Fig. 7).

In the critical regime, the shock is positioned between the lip and the throat. When the shock reaches the throat, the maximum possible total pressure recovery and the maximum mass flow recovery are obtained. The maximum mass flow recovery can be calculated from Eq. 15 if the Mach number variation and the distribution of the perforations are known. As a first approximation, the Mach number variation can be assumed to be that of a reversed de Laval nozzle. The maximum possible total pressure recovery as based on the normal shock theory is then equal to the total pressure ratio across a normal shock at the throat Mach number which can be determined from the maximum mass flow recovery (Eq. 9). If the shock takes up any other position in the convergent portion of the diffuser, the flow was found to be oscillatory (Ref. 2). A possible explanation of the flow oscillation may be ascribed to the normal shock bifurcation at the boundary layer (Ref. 13 and 14). At the triple point of the Mach configuration, the slip line of the vortex sheet divides the flow field into regions in which the static pressures are equal on both sides but the velocities are different (subsonic at the Mach stem). As a consequence, the large pressure rise exerted on the boundary layer causes the flow to separate from the diffuser wall and reduces the combined mass flow through the throat and through the perforations downstream of the shock. The accumulation of mass flow downstream of the shock is then believed to cause the shock to move toward the lip in view of the equation of continuity (Ref. 12). As soon as the shock is far enough away from the throat, the mass flow through the throat and the spillage downstream of the shock are increasing so that the shock advances again toward the throat. Hence, the shock oscillates in order to adjust the mass flow variation downstream of the shock due to flow separation. As a whole, the origin and mechanism of flow oscillation in a perforated diffuser is not completely understood.
The experimental results from Ref. 2 (carried out a free-stream Mach number of 1.90) were used in Fig. 7 for a comparison with theoretical values. Figure 7 shows the total pressure recovery versus the mass flow recovery. In the subcritical regime, the theoretical total pressure recovery is 0.767 which is equal to the total pressure ratio across a normal shock at the initial Mach number of 1.90 and the mass flow recovery varies from 0 up to 0.767. This regime is indicated as line A-B. For the supercritical operation, the theoretical maximum mass flow recovery is 0.962 from which the theoretical throat Mach number is found to be 1.23 (Eq. 9). Then, the theoretical peak total pressure recovery is 0.990 based on a normal shock at this throat Mach number. Line C-D indicates this supercritical regime. From lines A-B and C-D, the critical regime is determined as line B-C. It can be seen in Fig. 7 that the experimental points closely follow lines A-B and C-D except at point C where the experimental peak total pressure recovery is 0.909 as compared with the theoretical value of 0.990. In the critical regime, no readings could be taken as the flow in the diffuser was oscillatory. This shows that this instability occurs when the curve of total pressure recovery versus mass flow recovery has a positive slope of a sufficiently large magnitude (see line B-C in Fig. 7 and Refs. 14 and 15).

The above experimental results are good for a perforation ratio of 0.630. That the perforation ratio affects the flow characteristics is also shown in Fig. 7. At the perforation ratio of 0.570, the subcritical regime remains the same as above (line A-B) but there is no well-defined critical regime which indicates that the bow shock is not swallowed. As the flow expands supersonically downstream of the throat another shock occurs in the divergent portion. This time the supercritical regimes prevails showing line B-E of which the theoretical maximum mass flow recovery is 0.767 (equal to the total pressure recovery in the subcritical regime). Figure 7 shows that the diffuser at this perforation ratio (0.570) is under-perforated and that the experimental values coincide with the theoretical line A-B-E. For the third perforation ratio of 1.050, the diffuser must be considered over-perforated. The experimental curve at this perforation ratio indicates that the flow in both the subcritical and critical regimes is oscillatory (Fig. 7). The total pressure recovery in the subcritical regime is double-valued; about 0.767 for steady flow and about 0.834 for oscillating flow. Because of over-perforation, the maximum mass flow recovery reduces to 0.908 as compared with 0.962 at the perforation ratio of 0.630. It can be seen in Fig. 7 that this experimental curve deviates considerably from the theoretical line (A-B-C-D).

In evaluating the usefulness of the theoretical line A-B-C-D, it can be said that if the experimental points are far off this line, the diffuser is either under- or over-perforated.
III. APPARATUS AND PROCEDURE

3.1 Supersonic Wind Tunnel

The investigation was conducted at a free-stream Mach number of 2.50 and a Reynolds number of $2.50 \times 10^5$ per inch in the UTIA 16 inch by 16 inch intermittent blow-down supersonic wind tunnel (Ref. 17). From a wind tunnel calibration, the supersonic flow in the test section was found to be uniform except in the region close to the windows in the side walls. The flow in the empty test section as recorded on a schlieren photograph (Plate 3A) has a disturbance originating from the side of the lower block of the supersonic nozzle. It was probably caused by a small bump in the block. To verify this, an artificial trip (Scotch tape) was stuck near the bump and then another schlieren photograph was taken. As shown in Plate 3B, the Mach wave generated by the tape has the same intensity and inclination as the disturbance from the bump. This disturbance could probably be removed by re-surfacing the block. For the present model testing it was of no consequence.

The total pressure of the free-stream at the test section was assumed to be atmospheric in spite of the fact that the wind tunnel had two 30 mesh turbulent screens having an area of 5 feet by 5 feet. A total pressure loss of 0.13% of the free-stream total pressure was calculated from the data in Ref. 18, and it was considered to be negligible for this work.

3.2 Schlieren Photography

The schlieren system used in the present investigation is described in Ref. 17. Some improvements, however, were made during the wind tunnel testing. First, the mount of the BH-6 mercury vapor lamp was modified so that the lamp was easily accessible for adjustment to vertical, horizontal or 45 degree knife-edge positions. Secondly, a graded filter, which is a photographic plate of the image of a glass wedge (Ref. 19) was used instead of a knife edge. The reason was that the position of the knife edge was very critical owing to vibrations from several sources. Even if the knife edge was positioned correctly, the image of the shock waves on the film was overloaded and not sharp as shown in Plate 4A as compared with Plate 4B which was taken under the same conditions with the graded filter. The clarity of the shock pattern in Plate 4B illustrates the improvement.

The exposure time for schlieren photography depends on whether the flow is steady or not. Plates 5A and 5B were taken under the same test condition but the exposure times were 2.5 milli-seconds and 5 to 10 micro-seconds respectively. The latter exposure time was obtained from a flash circuit for the mercury lamp (Ref. 17) but the system did not always flash at the desired instant. To make it reliable, a Tesla coil was used. The tip of the coil was placed close to the mercury lamp so that the coil, as controlled by a Wollensak Rapax shutter, supplied some pre-ionization to the lamp and triggered the flash positively.
In addition, a Fairchild high-speed motion picture camera (HS-100) was set up to investigate the unsteady flow in the two-dimensional diffuser. The film speed was approximately 2,000 frames per second and the exposure time was about 100 micro-seconds. The continuous light source used for this purpose was a Western Union 300 watt D.C. concentrated arc lamp.

3.3 Perforated Diffusers

A two-dimensional and an axisymmetric diffuser were originally designed and tested in Refs. 5 and 4 respectively. Plate 1 shows a typical view of these two models. The leading dimensions are given in Figs. 1 and 2. The two-dimensional model consisted of two steel nozzle blocks held by two parallel plates with mounted windows. For this investigation, the two-dimensional diffuser was modified as follows: three additional rows of perforations were drilled close to the throat; the downstream edges of the orifices were chamfered; and eleven holes were tapped downstream of the throat, that could be plugged by screws for convenience of installing pressure probes to traverse along the longitudinal axis of the diffuser.

The axisymmetric diffuser was made of steel. This diffuser was also modified by: inserting a 6 inch long pipe, having the diameter of the throat (1.23 inches), between the convergent and divergent portions of the diffuser; countersinking the holes of every second row from the outside of the diffuser; adding one more row of perforations closest to the throat.

3.4 Pressure Measurements

Instrumentation of the two-dimensional diffuser consisted of nineteen static pressure taps on the centerline of one side window and seven pitot tubes mounted on the exit plug for total pressure measurements at the measuring plane. On the axisymmetric diffuser, there were eighteen static pressure taps along the diffuser wall and five pitot tubes at the measuring plane.

Cone, pitot, and static probes as shown in Plate 2 were designed and made for measuring the flow properties along the longitudinal axis of the two-dimensional diffuser. The cone probes having a semi-angle of 20 degrees with four surface pressure taps leading to a common tube were used in conjunction with pitot tubes to determine the total pressure and the static pressure at a point in the diffuser (Fig. 8 and Ref. 20). The cone probe could not be employed in the vicinity of the throat due to its comparatively large size but it was useful for checking at other points. Conventional static tubes, round-nosed and conical-nosed, were tried. One of the conical-nosed static tube, having an outside diameter of 0.057 in., was found to be accurate in a calibration test. This probe and a pitot tube
of the same size were used to determine the total pressure at any point in the flow (Ref. 21). The error in pressure readings obtained from the manometer was approximately 2%, causing an error in the calculated total pressure of about 5%.

All probes could be installed along the axis of the convergent and divergent portions of the two-dimensional diffuser. The size of the probes was small enough to pass through the perforations or tapped holes downstream of the throat. The stem of the probes were held by an adapter, which could be fastened to the web of the diffuser.

3.5 Experimental Procedure

Many schlieren photographs were taken of the flow about the two-dimensional diffuser with the graded filter set horizontally, vertically, and inclined at 45 degrees at an exposure time of 5 to 10 micro-seconds. The exit ratio (plug exit area divided by throat area) was varied from zero up to 1.65. The perforation ratios of the two-dimensional diffuser were progressively increased from 3.65 to 4.16 and 4.27. At the last ratio, the schlieren pictures showed that the shock was definitely positioned downstream of the throat. The cone, pitot, and static probes as well as the wall static pressure taps were then used to determine the flow properties in the diffuser. Detailed pressure measurements from point to point along the longitudinal axis of the two-dimensional diffuser were conducted only at three exit ratio settings (1.65, 1.34 and 0.824) and at a perforation ratio of 4.27.

Conclusions drawn from the above tests on the two-dimensional diffuser were then applied to improve the total pressure recovery and the mass flow recovery of the axisymmetric diffuser. The following tests were carried out: (1) A long throat, i.e. a 6 inch long pipe, was inserted between the convergent and divergent portions of the diffuser. The total pressure recovery versus the mass flow recovery were recorded and compared with those obtained without the long throat. (2) The straight holes of the perforations were countersunk from the outside of the diffuser to give a sharp edge. The wall static pressures were measured and compared. (3) The effect of closing some rows of perforations on the wall static pressure was investigated. First, holes closest to the lip were plugged row after row with Scotch tape until the wall static pressures started to change. Then, the same procedure was repeated starting with the row closest to the throat. (4) An extra row of perforations next to the throat was added and the effect on the wall static pressure along the diffuser was observed.
IV. RESULTS AND DISCUSSIONS

4.1 Two-Dimensional Diffuser

4.1.1 Schlieren Photographs

In Ref. 5, the two-dimensional diffuser with the perforation ratio of 3.65 was tested with the distribution of holes as shown in Fig. 9. In the present investigation, a series of schlieren photographs were taken with the same perforation ratio at different plug openings. A few of these were selected for discussion. Plate 6A illustrates the flow pattern about the diffuser when the plug was fully closed. It can be seen that no bow shock appears upstream of the lip. Plates 6B, 6C and 6D were taken at an exit ratio of 0.824 with the graded filter placed horizontally, vertically, and inclined at 45 degrees respectively. These plates show that a prominent lambda shock occurs in the convergent portion of the diffuser. The lambda shock is a normal shock bifurcated by the boundary layer. Downstream of the shock, the flow is not entirely subsonic and the shock pattern is very irregular. Outside of the diffuser, the second oblique shock—downstream of the first oblique shock originating from the lip—is caused by the spillage through the perforations. The position of the second external oblique shock and the location of the internal lambda shock are connected in a way similar as are the centre and curved part of a bow shock. These three photographs also show that the lambda shock is located at different positions, suggesting that the flow is nonstationary (oscillating) in the diffuser.

Plate 6E taken at an exit ratio of 1.24 exhibits two shock systems in the diffuser: the lambda shock in the convergent portion and the multiple shocks in the divergent portion. This means that this diffuser at the perforation ratio of 3.65 has still not swallowed the shock. With the plug wide open (exit ratio = 1.65), Plate 6F shows that the wave pattern upstream and in the vicinity of the throat is very irregular which demonstrates that the shock cannot be swallowed. In the divergent portion, the multiple shocks are considerably downstream of the throat since the position of the multiple shocks can be controlled by the plug opening.

This series of schlieren photographs indicates that the internal shock in the diffuser does not follow the pattern as discussed in Sec. 2.5. In the theoretical analysis, the bow shock in the subcritical regime is first detached from the lip and then attaches when the plug is opened from a closed position. In the supercritical regime, the shock is swallowed and appearing only in the divergent portion. However, Plates 6A and 6F do not satisfy these conditions for possibly two reasons: the diffuser at the perforation ratio of 3.65 seems to be over-perforated and the perforated area close to the throat is not sufficient. The perforation distribution and the perforation ratio are two of the main factors to complete the shock swallowing process. These two parameters can be changed either by closing off or by adding perforations. As it was not easy to close perforations on the two-dimensional model, the axisymmetric diffuser was used for this investigation reported on in Sec. 4.2.
Perforation Ratio Equal to 4.16

Two rows of perforations close to the throat were added to increase the perforation ratio to 4.16. The distribution of perforations is shown in Fig. 9. When the plug was fully closed, Plate 7A shows that there is another normal shock downstream of the lambda shock. In Plates 7B and 7C, the two shock system similar to that in Plates 6D and 6E appears again. Plate 7D, taken at the exit ratio of 1.65, shows that the shock in the convergent portion has been swallowed. However, the waves in the vicinity of the throat are not sharp and indicate some unsteadiness although the exposure time for the photographic plate was 5 to 10 micro-seconds only.

Perforation Ratio Equal to 4.27

At the perforation ratio of 4.27, Plates 8 were taken at different plug openings. Plate 8A clearly shows the slip lines of the lambda shock or shocks. Plates 8B to 8G are similar to those noted for the previous perforation ratio. In Plates 8F and 8G, no lambda shock occurs in the convergent portion of the diffuser and all waves are sharply criss-crossed. This provides convincing evidence that the shock has been swallowed.

High-Speed Schlieren Motion Pictures

To study the shock oscillation in detail, a Fairchild motion picture camera (about 2,000 frames per second) was employed to take schlieren photographs. Plates 9B to 9S (18 frames) represent a complete cyclic motion (about 110 cycles per second) of the shock at the perforation ratio of 4.27 and the exit ratio of 0.824 which are the same as those for Plates 8B to 8E. The internal shock pattern in the convergent portion in Plates 9 (exposure time about 100 micro-seconds) are not as clear as that in Plates 8B to 8E (exposure time 5 to 10 micro-seconds). Since the internal shock and the second oblique shock downstream of the first oblique shock originating from the lip are connected as one unit as discussed before (Plates 6B to 6D), the distance between these oblique shocks was used to indicate the shock oscillation. As shown in Plates 9, there is no normal distance between them, but a third oblique shock emanating from the joint of the nozzle block in the wind tunnel test section intersects these two external oblique shocks. The distance between the first and second oblique shocks along the third one was measured as shown in Fig. 10 in which the relative displacement (distance measured divided by maximum distance) is plotted as a function of frame number. Figure 10 shows that from Plate 9B to 9M (12 frames) the second oblique shock moves from the position closest to the lip to that closest to the throat, and that from Plates 9M to 9T - the reverse process takes place.
If Plates 9 are examined closely, it can be seen that from Plates 9H to 90 (8 frames) the lambda shock remains very close to the throat. This suggests that the flow in the diffuser during this time interval (Plates 9H to 90) can be considered to be in equilibrium. The flow pattern in the receding process (Plates 9P to 9S) is very irregular and chaotic. This means that the equilibrium state is overthrown suddenly and this is possibly caused by the interaction of shock waves with the boundary layer through which the pressure disturbances travel upstream to affect all the waves inside and outside the diffuser as shown, e.g., in Plates 9P to 9S (4 frames).

4.1.2 Total Pressure Recovery and Mass Flow Recovery

The total pressure recovery was obtained from 7 pitot tubes mounted on the exit plug. The mass flow recovery was determined from the total pressure recovery and the exit ratio by applying Eq. 7. The total pressure recovery versus the mass flow recovery is shown in Fig. 11 for perforation ratios varied from 1.72 up to 4.27 at exit ratios ranging from 0 to 1.65. At the perforation ratio of 4.16 (not 4.27), the highest value of the maximum mass flow recovery was found experimentally to be 0.816 and the corresponding peak total pressure recovery (at the same perforation ratio) was 0.570. From this maximum mass flow recovery, the theoretical throat Mach number becomes 1.57 (see Eq. 9 and Fig. 3) from which the theoretical peak total pressure recovery of 0.906 - based on the normal shock theory - is obtained. In Fig. 11, the theoretical line A-B-C-D is plotted for comparison. The discrepancy of 0.336 in peak total pressure recovery will be discussed in Sec. 4.1.3.

Figure 12 shows the maximum mass flow recovery as a function of the perforation ratio. When this ratio increases from 0. to 1.72, the maximum mass flow recovery is approximately constant. At the perforation ratio of 1.72, the bow shock becomes for the first time attached to the lip of the diffuser as shown in Ref. 5. It follows that 1.72 is the minimum perforation ratio for shock attachment, from which the subsonic flow coefficient was calculated to be 0.581 (Eq. 11) as compared with the theoretical coefficient of contraction of 0.64 (see Sec. 2.2). When the perforation ratio increases from 1.72 to 3.65, the maximum mass flow recovery increases gradually. However, a further increase from 3.65 to 4.16 and 4.27 causes the maximum mass flow recovery to jump from 0.712 to 0.816 and then sharply to drop to 0.730. This sudden change in the mass flow recovery makes the last three experimental points in Fig. 11 rather dubious.

For the perforation ratio of 4.16 at which the highest value of the maximum mass flow recovery was obtained, Figs. 11 and 12 do not signify that this is the perforation ratio required to complete the shock swallowing process. It was even suspected that the diffuser was overperforated and that the perforations close the lip were superfluous. If this
is so, closing rows of perforations close to the lip of an over-perforated diffuser should not affect the shock swallowing process. As it was not very easy to verify this point on the two-dimensional diffuser, the axisymmetric diffuser was used. The findings are included in Sec. 4.2.2.

4.1.3 Pressure Measurement

The total pressure losses in a perforated diffuser are probably caused by the lambda shock in the convergent portion, the multiple shocks in the divergent portion, the internal oblique shocks originating from the perforations, by wall friction and flow separation caused by the interaction of the shock waves with the boundary layer. In order to analyse these losses, the total pressures along the longitudinal axis of the two-dimensional diffuser were measured at the perforation ratio of 4.27.

In this report, isentropic lines are used as reference values for the experimental results. They are defined as the distribution of total, pitot, and static pressures, and Mach numbers along the axis of a reversed de Laval nozzle. In the convergent portion, the isentropic lines are based on the area ratios and the throat Mach number of unity. In the divergent portion, isentropic lines were determined from area ratios and throat Mach numbers which were found by trial and error such as to fit the experimental points.

Exit Ratio Equal to 1.65

In Fig. 13, the experimental points of total, pitot, and static pressures measured along the axis of the two-dimensional diffuser are plotted together with the above-defined isentropic lines. The total pressures measured along the diffuser axis were found to be approximately equal to the free-stream total pressure until a linear reduction of total pressure takes place in a region downstream of the throat. The total pressure loss due to multiple shocks occurs over a distance of 4.5 throat heights downstream of the throat. Further downstream of this region, the total pressure seems to reach another constant level at 0.275 of the free-stream total pressure. Since the Mach number close to and upstream of the multiple shocks was found to be 2.33, the maximum possible total pressure recovery would be 0.570. The discrepancy of 0.295 in total pressure recovery is probably due to flow separation caused by the interaction of shock waves with the boundary layer. To further discuss this discrepancy, an understanding of multiple shocks in pipe flow is necessary. It is known that the transition from supersonic to subsonic flow in a pipe takes place across a series of shocks or multiple shocks (Ref. 9). The extent of the multiple shock region was found to vary between 14 diameters for a Mach number of 4.2, and 9 diameters for a Mach number of 1.8. Although the multiple shocks in pipe flows are usually distorted and are far from being normal shocks, the overall change in fluid properties is in good agreement with that for a single normal shock wave. This shows that the loss due to flow separation must
be relatively small in the pipe. However, for the divergent portion of a diffuser, not only the shocks but also the included angle impose a large adverse pressure gradient along the wall. The resulting flow separation causes an additional total pressure loss as illustrated by the above discrepancy. The separation loss for the present model could probably be reduced if the included angle would be less than 7 degrees and if the diffuser could be tested at higher Reynolds numbers. In addition, a long throat coupled to a normal shock diffuser was found (Ref. 3) to be successful in increasing the total pressure recovery, but it was not when coupled to our axisymmetric perforated diffuser as reported in Sec. 4.2.1.

The static pressures measured in the convergent portion either along the diffuser axis or on the diffuser wall agree with the isentropic line except in the region close to the throat where the pressures are below the isentropic line (Fig. 13). The deviation of measured static pressures is due to spillage as indicated by Eq. 1. Since the total pressure at any section in the convergent portion of the diffuser is equal to the free-stream total pressure, the spillage reduces the Fliegner number at a certain cross-sectional area so that the local Mach number becomes higher than its isentropic value (see Fig. 3) which in turn reduces the static pressure to a value lower than its isentropic value (Eq. 1). On the other hand, in the divergent portion, the measured static pressures coincide with another isentropic line corresponding to a throat Mach number of 1.70 except in the shock region where the static pressures are above this isentropic line. In this shock region, the wall static pressures are higher than those along the diffuser axis because of the shock-boundary-layer interaction.

In Fig. 14, the variation of Mach number along the diffuser is compared with the isentropic lines. These Mach numbers were obtained by three methods: from pitot and static pressure measurements, from wall static and free-stream total pressures, and from the angles of waves emanating from the perforations as measured off from an enlarged schlieren photograph (Plate 8G). These three distributions follow closely the isentropic lines (see Fig. 14) except in the vicinity of the throat and in the region where the multiple shocks occur. Since the wave angle measurements of Mach number are in substantial agreement with the pitot-static method, it is reasonable to say that these waves are truly Mach waves.

Besides for the determination of the maximum possible total pressure recovery, the wall static pressures were also found useful in determining the maximum mass flow recovery. From the wall static pressure measurements and the corresponding isentropic line in the divergent portion, the throat Mach number was determined to be 1.70 which was used to find the maximum mass flow recovery of 0.716 (Eq. 9). Furthermore, by using the Mach number variation obtained from wall static pressures in conjunction with supersonic flow coefficients, the maximum mass flow recovery was calculated to be 0.73 (Figs. 4, 9 and 14 and Eq. 14).
The experimental value based on the total pressure measured at the measuring plane was found to be 0.73 also (Fig. 11). Judging from the close agreement of these results, it appears that all three methods are valid.

Exit Ratio Equal to 1.34

Since the multiple shocks in the diffuser at the exit ratio of 1.34 are still downstream of the throat as shown in Plate 8F, Figs. 15 and 16 respectively show the pressure distribution and Mach number variation along the diffuser axis to be similar to those of the previous case. The total pressure measured at a position downstream of the multiple shocks is 0.512 of the free-stream total pressure, whereas the theoretical peak total pressure recovery, as calculated from the Mach number of 1.82, is 0.804. This Mach number was measured at a position close to and upstream of the multiple shocks (see Fig. 16). This discrepancy of 0.292 remains approximately the same as in the previous case. No reasons can be given why the additional total pressure loss due to flow separation is independent of the shock Mach number or its position.

Exit Ratio Equal to 0.824

The total pressure distribution and the Mach number variation along the diffuser axis as shown in Figs. 17 and 18, respectively, disclose that the lambda shock occurs now in the convergent portion of the diffuser. In Fig. 17, the total pressure decreases monotonically in the divergent portion in similarity to the previous cases. However, it does no longer decrease monotonically in the convergent portion where the lambda shock appears as shown in Plates 8B to 8E. The fluctuation of total pressure is possibly due to strong mixing behind the oscillating lambda shock. As to the static pressure distributions, the interaction of the shock with the boundary layer causes the wall static pressures to rise considerably above the isentropic line in the convergent portion. The wall static pressures in the convergent portion are a little lower than the static pressure measured along the diffuser axis; this is due to the spillage. In the divergent portion, because of the flow separation, the wall static pressures are a little higher than the pressures measured along the diffuser axis.

In Fig. 18, the Mach number variation in the convergent portion shows that across the lambda shock the Mach number does not drop from supersonic to subsonic. In the vicinity of the throat, the flow speeds up supersonically. Downstream of the throat, the flow becomes transonic and then subsonic. The Mach number was further determined in the convergent portion by measuring the cone wave angle of a cone probe. These cone wave angles were determined from the photographs of Plates 10A to 10D which were enlarged more than 10 times. The Mach number variation from the cone wave angle measurements shows the same trend as that obtained from the pressure measurements. Furthermore, these
photographs show that the flow pattern in the diffuser tend to remain similar whereas on Plates 8B to 8E changes are noted. This suggests that a body introduced into the convergent portion may tend to stabilize the flow in the diffuser.

**Test Results from Refs. 1 and 2**

The test results from Refs. 1 and 2 were used to check whether the wall static pressures in the divergent portion of a diffuser can be used to determine the throat Mach number and the maximum mass flow recovery. In Fig. 19, the wall static pressures along the divergent portion of the diffuser of Ref. 1 were replotted together with two isentropic lines. The wall static pressures measured at a free-stream Mach number of 1.85 coincide with the isentropic line having a throat Mach number of 1.20, from which the maximum mass flow recovery was found (from Eq. 9 and Fig. 3) to be 0.970. The experimental maximum mass flow recovery is not given in Ref. 1. In Ref. 2, however, a diffuser having the same contraction ratio as in Ref. 1 was tested in the same wind tunnel at a free-stream Mach number of 1.90. The maximum mass flow recovery was determined to be 0.975. Judging from the close agreement of the above results and from the results of this report (see Sec. 4.1.3 under Exit Ratio Equal to 1.65), it is reasonable to assume that the wall static pressure distribution in the divergent portion can be used to find the actual throat Mach number and then the maximum mass flow recovery.

In Ref. 1, the peak total pressure recovery was measured to be 0.931 but the maximum possible total pressure recovery is 0.993, based on the theoretical throat Mach number of 1.20. The discrepancy of 0.062 (about 6 percent) is small. But it should be kept in mind that the diffuser was tested at the comparatively low free-stream Mach number of 1.85 and that it had a small included angle of 5 degrees in the divergent portion which would help to reduce the loss due to flow separation.

**4.2 Axisymmetric Diffuser**

**4.2.1 Throat Length**

The axisymmetric diffuser originally had a perforation ratio of 4.28 as shown in Fig. 20. To begin with, this diffuser was tested to obtain the total pressure recovery as a function of the exit ratio and the mass flow recovery. The results are shown in Fig. 21, including the theoretical characteristic lines A-B-C-D and A-B-E. They are similar to those of Fig. 11 for the two-dimensional diffuser. The maximum mass flow recovery was found to be 0.880 from which the theoretical throat Mach number follows as 1.40 (Eq. 9 and Fig. 3). Based on this throat Mach number, the maximum possible pressure recovery is 0.958 as compared with the measured total pressure recovery of 0.778.
In an attempt to increase the measured peak total pressure recovery, a long throat, as discussed in Ref. 3, was thought to be useful. The long throat was a 6 inch long pipe having a diameter of 1.23 inches, the same as the throat diameter of the diffuser. With this long throat, the peak total pressure recovery measured was 0.790 (see Fig. 21). Such a small increment of total pressure recovery, it seems, would not justify the use of a long throat. This test showed further that it was not possible to confine the multiple shocks to the throat alone in order to avoid the loss due to flow separation in the divergent portion of the diffuser.

As shown in Fig. 21, the maximum mass flow recovery in the supercritical regime does not remain constant as expected but increases as the total pressure recovery decreases. This is contradictory to not only the conclusion in Sec. 2.4 but also to the experimental results of Ref. 2. Since the maximum mass flow recovery was obtained from the total pressure recovery by use of Eq. 7, a correct determination of the average total pressure at the measuring plane requires a number of pitot tubes. In this axisymmetric diffuser, 5 pitot tubes, installed at the measuring plane of 2.25 inch diameter, seem not enough and are far less than the 40 pitot tubes installed in the measuring section of 3.60 inch diameter in the diffuser of Ref. 2. Figure 21 indicates that there is an approximately 15% increase in the maximum mass flow recovery which is believed to be the result of an inadequate measurement of the average total pressure.

4.2.2 Perforations

As discussed in Sec. 2.2, the subsonic flow coefficient depends on the shape of the perforation hole. Perforations which are countersunk from the outside of the diffuser are more effective than straight holes. Similarly, the supersonic flow coefficient is affected by the orifice shape. Because the thickness of the diffuser wall is reduced by countersinking the perforations, the supersonic flow at the perforation exit can easily spill out without blocking effects due to the wall thickness. Figure 22 shows the effect of countersinking at a perforation ratio of 4.28 and an exit ratio of 2.50 for the axisymmetric diffuser, as a reduction in wall static pressures in the vicinity of the throat because of the higher spillage (Eq. 1). It can be said that countersunk holes with sharp edges have the same effect as an increase in the perforated area.

Figure 23 shows the change in wall static pressures caused by closing off rows of perforations. When the first 8 rows of perforations (out of 16) at the lip of the diffuser were closed with Scotch tape, very small changes in wall static pressures were noted (see points marked X in Fig. 23). This suggests that closing these 8 rows of perforations which reduced the perforation ratio from 4.28 to 1.69 had a negligible effect on the flow in the diffuser. With a subsonic flow
coefficient of 0.581, which was experimentally determined in Sec. 4.1.2, the minimum perforation ratio is 1.72 (Eq. 12). This shows that the perforation ratio required to swallow the shock is approximately the same as the minimum perforation ratio. Next, when the last row (No. 16) of perforations out of 16 close to the throat was closed, the wall static pressures were raised well above the isentropic line in the convergent portion of the diffuser (see points marked Δ in Fig. 23). This means that the most effective location for perforations is in the region close to the throat where the static pressure difference across perforations is large.

In Sec. 4.1.3, it has been concluded that if the shock is swallowed, the wall static pressures in the convergent portion of the two-dimensional diffuser agree with the isentropic line except in the region close to the throat where the pressures are below the isentropic line (Fig. 13). In testing the axisymmetric diffuser previously (Ref. 4), it has been assumed that the shock was swallowed because the peak total pressure recovery was obtained. The same model without any modification was re-tested (see points marked + in Fig. 23). It can be seen that the wall static pressures close to the throat in the convergent portion are still a little above the isentropic line. By adding one row of perforations (No. 17) next to the throat, the wall static pressure are reduced below the isentropic line (see points marked D in Fig. 23). However, no further tests were attempted to prove that the extra row should increase the total pressure recovery.

V. CONCLUSIONS

The following conclusions may be drawn from the present investigation: (1) By introducing the Fliegner number as a flow parameter it has been possible to simplify all the theoretical derivations and calculations given in Refs. 1 and 2. The mass flow recovery and the total pressure recovery can be simply related to the Fliegner number.

(2) Schlieren photographs have shown that a perforated diffuser can swallow the shock. However, the concept of the shock swallowing process based on a single normal shock is a much oversimplified model. Actually, the normal shock is modified by the boundary layer and becomes a lambda shock. At the same time a system of multiple shocks occupy a region of 4 to 10 throat heights in the divergent portion of the two-dimensional diffuser. A Fairchild high-speed motion picture camera has shown that not only the lambda shock in the convergent portion, but also the entire internal and external flows bounded by the shocks, oscillate along the longitudinal axis of the present two-dimensional diffuser when it undergoes subcritical operation. The frequency of oscillation was 110 cycles per second at the perforation ratio of 4.28 and the exit ratio of 0.824. The shock oscillation in Plates 8B to 8E, it seems, can be eliminated by introducing a small body into the convergent portion as shown in Plates 10A to 10D but as yet no explanation can be given of this phenomenon.
The total pressure measurements along the two-dimensional diffuser axis have shown that there were no total pressure losses (within the experimental error) upstream of the multiple shocks when the main shock was swallowed. The oblique disturbances originating from the perforations were effectively Mach waves. The loss of total pressure, however, was more than that of a normal shock in the divergent portion by about 30% of the free-stream total pressure. This probably arises from flow separation due to the interaction of the multiple shock waves with the boundary layer in the divergent part. The wall static pressures upstream of the multiple shocks agreed with the static pressure measured by a static tube along the two-dimensional diffuser axis confirming that one-dimensional flow theory applies. Consequently, it is not necessary to probe this type of flow since its properties can be calculated from the wall static pressures. The wall static pressures are also useful to indicate the presence of shocks in the convergent portion of a diffuser. If the wall static pressures in the convergent portion were below the isentropic line (Fig. 13), the shock was swallowed (Plate 8G) and vice versa (Fig. 17 and Plates 8B to 8E). Furthermore, the wall static pressures in the divergent portion in conjunction with the isentropic line can also be employed to determine the throat Mach number from which the maximum possible total pressure recovery and the maximum mass flow recovery can be calculated (see Sec. 4.1.3 under Test Results from Refs. 1 and 2). To sum up, the wall static pressures together with the isentropic lines can provide a simple and reliable method of investigating perforated diffusers.

A long pipe inserted between the convergent and divergent portions of the axisymmetric diffuser was not as effective as expected in increasing the peak total pressure recovery. This means that the multiple shocks could not be confined to the long throat alone as they were according to the experimental results of Ref. 3.

Countersinking of the perforations from the outside of the diffuser had the same effect on the flow properties as an increase in perforated area. Perforations close to the lip of the axisymmetric diffuser were much less effective than those close to the throat. As a result, the perforation ratio could be reduced from 4.28 to 1.69 without changing the wall static pressures. The minimum perforation ratio becomes 1.72 (Eq. 12) if a subsonic coefficient of 0.581, as experimentally determined for the two-dimensional diffuser, is used. The best location for perforations was found to be in the region close to the throat where the static pressure difference across the perforations was larger.

Finally, the perforated diffuser has still two problems that must be solved before it can be used effectively in practice: (a) the subsonic diffuser should be designed so as to avoid flow separation; (b) the external drag due to spillage must be reduced. Both problems, although recognized in the present work, were not investigated.
APPENDIX

ESTIMATION OF THE SUPersonic FLOW COEFFICIENT

The spillage through the perforations in the convergent portion of a diffuser may be estimated in the supersonic region by means of the Prandtl-Meyer flow around a corner. It is assumed that the static and total pressures at the perforation exit are the same as the free-stream pressures. At a certain station in the diffuser, the flow having a local Mach number \((M_x)\) at the free-stream total pressure \((P_0)\) will turn at the upstream edge of the perforation hole until the Mach number at the perforation exit \((M_p)\) is equal to the main free-stream Mach number \((M_0)\). The following sketch shows that the flow through the perforation is at the free-stream Mach number, and a shock wave is formed as the flow turns parallel to the diffuser wall.

Sk. 1. Model of a Supersonic Flow Through a Perforation Hole
Since $M_p = M_0$, then $\alpha_p = \alpha_0$ and $\omega_p = \omega_0$. From Eq. 1, the following equations can be derived:

$$F(M_p) = F(M_0) = \frac{4 \Delta m_p \sqrt{T_0}}{\pi \sqrt{R \gamma}}$$  \hspace{1cm} (A1)

and

$$F(M_x) = \frac{4 \Delta m_p \sqrt{T_0}}{\pi \sqrt{R \gamma \gamma'}}$$  \hspace{1cm} (A2)

dependent

therefore

$$\frac{y_x}{y_y} = \frac{F(M_0)}{F(M_x)}$$  \hspace{1cm} (A3)

and from geometric relations follows (see Fig. SK. 1)

$$y_y = \gamma \sin(\omega_p - \omega_x) = \gamma \sin(\omega_0 - \omega_x)$$  \hspace{1cm} (A4)

By definition, the supersonic flow coefficient, $Q_b$, is the ratio of the effective (elliptic) area to the geometric (circular) area of the perforation. Since the effective area is the flow area normal to the streamlines at $M_x$, the $Q_b$ is obtained from Eqs. A3 and A4 as,

$$Q_b = \frac{\pi y_x r}{\pi y^2} = \frac{F(M_0)}{F(M_x)} \sin(\omega_0 - \omega_x)$$  \hspace{1cm} (A5)

In Refs. 1 and 2, the same $Q_b$ was derived in a different way and expressed in a different form.

If the flow spilled out through the perforation cannot flow in a direction opposite to the main free-stream, the turning angle of the final Mach line is limited to a position parallel to the diffuser wall (Ref. 2). In this case, the angle between the final Mach line and the diffuser wall as shown in the sketch becomes zero, i.e.,

$$\omega_p - \omega_x = 0$$  \hspace{1cm} (A6)

Then, the supersonic flow coefficient involving this limiting condition is given as

$$Q_b = \frac{F(M_p)}{F(M_x)} \sin \omega_p$$  \hspace{1cm} (A7)
where $M_p$ is less than the free-stream Mach number $M_o$. The above equation can be solved by trial and error or graphically. For example, at a free-stream Mach number of 2.50, Eq. A7 becomes effective when $M_x$ is less than 1.62. In other words, Eq. A5 is valid when $M_x$ is higher than 1.62 as shown in Fig. 4. If $M_x = 1.10$, from Eq. A6, $M_p$ is found to be 2.12 instead of 2.50, the free-stream Mach number.
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FIG. 1 SCHEMATIC DRAWING OF TWO-DIMENSIONAL DIFFUSER (REF. 5)
MEASURING PLANE

EXIT PLUG

FIG. 2  SCHEMATIC DRAWING OF AXISYMMETRIC DIFFUSER (REF. 4)
FIG. 3 FLIEGNER NO. VERSUS MACH NO.
FIG. 4 VARIATION OF SUPERSONIC FLOW COEFFICIENT WITH MACH NUMBER.
FIG. 5 INTERNAL CONTOURS OF REVERSED DE LAVAL NOZZLES
FIG. 6 DISTRIBUTIONS OF STATIC PRESSURES ALONG REVERSED DE LAVAL NOZZLES
EXPERIMENTAL RESULTS
FROM NACA RM E50B02

FREE STREAM MACH NO. = 1.90
CONTRACTION RATIO = 1.55

PERFORATED AREA = \{ 0.630 FOR
THROAT AREA = 1.050 FOR

STEADY FLOW
OSCILLATING FLOW
IDEAL FLOW

FIG. 7 VARIATION OF OVERALL TOTAL PRESSURE RECOVERY WITH MASS FLOW RECOVERY. EXPERIMENTAL RESULTS FROM NACA RM E50B02 (REF. 2). FREE-STREAM MACH NUMBER = 1.90.
FIG. 8 VARIATION OF CONE SURFACE STATIC PRESSURE TO PITOT PRESSURE WITH MACH NO.  CONE SEMI-ANGLE = 20°
FIG. 9 DISTRIBUTION OF PERFORATIONS FOR TWO-DIMENSIONAL DIFFUSER
Fig. 10 Relative displacement of second oblique shock outside 2-D diffuser. Perforation ratio = 4.27, exit ratio = 0.824, frequency = 110 cyc/sec. Complete cycle from B to T.
Flow Characteristics
Two-Dimensional Diffuser

Exit Area / Throat Area
0.405 / 0.824

Perforated Area / Throat Area
4.27
4.16
3.65
2.43
1.72

Overall Total Pressure Recovery

Mass Flow Recovery

Actual Flow
Ideal Flow

Variation of Overall Total Pressure Recovery with Mass Flow Recovery for Two-Dimensional Diffuser.

FIG. 11
FIG. 12 VARIATION OF MAXIMUM MASS FLOW RECOVERY WITH PERFORATED AREA FOR 2-D. DIFFUSER.
PRESSURE MEASUREMENTS (2-D. DIFF., Ae/At=1.65)

- ISENTROPIC LINE (Mthroat = 1.0)
- ISENTROPIC LINE (Mthroat = 1.7)

D TOTAL P. FROM PITOT & CONE PROBES
X TOTAL P. FROM PITOT & STATIC TUBES
+ PITOT PRESSURE
A STATIC P. FROM PITOT & CONE PROBES
V STATIC P. FROM STATIC TUBE
O STATIC PRESSURE FROM WALL TAP

FIG. 13 DISTRIBUTION OF TOTAL, PITOT & STATIC PRESSURE ALONG TWO-DIMENSIONAL DIFFUSER.
PERFORATED TO THROAT AREA RATIO = 4.27, EXIT TO THROAT AREA RATIO = 1.65.
FIG. 14  VARIATION OF MACH NO. ALONG TWO-DIMENSIONAL DIFFUSER. PERFORATED TO THROAT AREA RATIO = 4.27, EXIT TO THROAT AREA RATIO = 1.65.
FIG. 15 DISTRIBUTION OF TOTAL, PITOT & STATIC PRESSURE ALONG TWO-DIMENSIONAL DIFFUSER. PERFORATED TO THROAT AREA RATIO = 4.27, EXIT TO THROAT AREA RATIO = 1.34.
FIG. 16 VARIATION OF MACH NUMBER ALONG TWO-DIMENSIONAL DIFFUSER. PERFORATED TO THROAT AREA RATIO = 4.27, EXIT TO THROAT AREA RATIO = 1.34.
PRESSURE MEASUREMENTS

2-D. DIFF., $\frac{A}{A_0} = 0.824$

--- ISENTROPIC LINE ($M_{throat} = 1.0$)
--- ISENTROPIC LINE ($M_{throat} = 1.7$)

D TOTAL PRESSURE FROM PITOT & CONE PROBES
X TOTAL PRESSURE FROM PITOT & STATIC TUBES
+ PITOT PRESSURE
Δ STATIC PRESSURE FROM PITOT & CONE PROBES
▽ STATIC PRESSURE FROM STATIC TUBE
O STATIC PRESSURE FROM WALL TAP

FIG. 17 DISTRIBUTION OF TOTAL, PITOT & STATIC PRESSURE ALONG TWO-DIMENSIONAL DIFFUSER. PERFORATED TO THROAT AREA RATIO = 4.27, EXIT TO THROAT AREA RATIO = 0.824.
MACH NUMBER VARIATION
2-D. DIFF., $\frac{Ae}{At} = 0.824$

FIG. 18 VARIATION OF MACH NUMBER ALONG TWO-DIMENSIONAL DIFFUSER. PERFORATED TO THROAT AREA RATIO = 4.27, EXIT TO THROAT AREA RATIO = 0.824.
EXPERIMENTAL RESULTS
FROM NACA TN 3767
(REF. 1)

--- ISENTROPIC LINE HAVING \( M_{\text{throat}} = 1.0 \)

--- ISENTROPIC LINE HAVING \( M_{\text{throat}} = 1.2 \)

\( \triangle \) EXPERIMENTAL POINT OF FIG. 4 (REF. 1)

FIG. 19 WALL STATIC PRESSURE ALONG A PERFORATED DIFFUSER. EXPERIMENTAL RESULTS FROM NACA TN 3767. FREE-STREAM MACH NO. = 1.85.
Figure 20: Distribution of perforations for axisymmetric diffuser.
FIG. 21 VARIATION OF OVERALL TOTAL PRESSURE RECOVERY WITH MASS FLOW RECOVERY FOR AXISYMMETRIC DIFFUSER SHOWING THE EFFECT OF THROAT LENGTH.
FIG. 22 WALL STATIC PRESSURE DISTRIBUTION ALONG AXISYMMETRIC DIFFUSER SHOWING THE EFFECT OF COUNTERSINKING.
FIG. 23 WALL STATIC PRESSURE ALONG AXISYMMETRIC DIFFUSER SHOWING THE EFFECT OF CLOSING PERFORATIONS.
PLATE 1  VIEW OF THE TWO-DIMENSIONAL (REF. 5) AND OF THE AXISYMMETRIC DIFFUSER (REF. 4)
PLATE 2  CONE PROBES (UPPER), STATIC TUBES (MIDDLE), AND PITOT TUBES (BOTTOM)
PLATE 3A SCHLIEREN PHOTOGRAPH OF SUPERSONIC FLOW (FROM RIGHT TO LEFT) IN EMPTY TEST SECTION. KNIFE-EDGE, HORIZONTAL; EXPOSURE, 2.5 MILL-SECONDS; PROMINENT MACH WAVE GENERATED FROM LOWER BLOCK.

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PLATE 4A  SCHLIEREN PHOTOGRAPH OF A DOUBLE WEDGE STATIC PROBE IN A SUPersonic FLOW. KNIFE-EDGE, HORIZONTAL; EXPOSURE, 2.5 MILLI-SECONDS.

PLATE 4B  AS IN 4A EXCEPT USING A GRADED FILTER INSTEAD OF KNIFE-EDGE.
PLATE 5A  SCHLIEREN PHOTOGRAPH OF FLOW PATTERN ABOUT A TWO-DIMENSIONAL DIFFUSER. GRADED FILTER, VERTICAL; EXPOSURE, 2.5 MILLI-SECONDS.
PLATE 5B
AS IN 5A EXCEPT EXPOSURE. 5 to 10 MICRO-SECONDS
PLATE 6A SCHLIEREN PHOTOGRAPH OF FLOW PATTERN ABOUT A TWO-DIMENSIONAL DIFFUSER. EXPOSURE, 5 to 10 MICRO-SECONDS; PERFORATED TO THROAT AREA RATIO, 3.65; EXIT, FULLY CLOSED; GRADED FILTER, VERTICAL
PLATE 6D AS IN 6B EXCEPT GRADED FILTER AT 45 DEGREES
PLATE 6E
AS IN 6A EXCEPT EXIT TO THROAT AREA RATIO = 1.24.
PLATE 6F
AS IN 6A EXCEPT EXIT TO THROAT AREA RATIO - 1.65.
PLATE 7A  SCHLIEREN PHOTOGRAPH OF FLOW PATTERN ABOUT A TWO-DIMENSIONAL DIFFUSER. EXPOSURE, 5 to 10 MICRO-SECONDS; PERFORATED TO THROAT AREA RATIO, 4.16; GRADED FILTERS; VERTICAL EXIT FULLY CLOSED.
PLATE 7B
AS IN 7A EXCEPT EXIT TO THROAT AREA RATIO = 0.824.
PLATE 7D
AS IN 7A EXCEPT EXIT TO THROAT AREA RATIO = 1.65.
PLATE 8A  SCHLIEREN PHOTOGRAPH OF FLOW PATTERN ABOUT A TWO-DIMENSIONAL DIFFUSER.  EXPOSURE, 5 to 10 MICRO-SECONDS; PERFORATED TO THROAT AREA RATIO, 4.27; GRADED FILTER, HORIZONTAL; EXIT, FULLY CLOSED.
PLATE 8B AS IN 8A EXCEPT EXIT TO THROAT AREA RATIO = 0.824.
SHOCK POSITION (S) INDICATED.
HIGH-SPEED SCHLIEREN MOTION PICTURES OF FLOW OSCILLATION IN A TWO-DIMENSIONAL DIFFUSER.
CAMERA SPEED, ABOUT 2,000 PICTURES PER SECOND; KNIFE-EDGE, VERTICAL; EXPOSURE, ABOUT 100 MILLI-SECONDS; PERFORATED TO THROAT AREA RATIO, 4.27; EXIT TO THROAT AREA RATIO, 0.824; AVERAGE FREQUENCY, 110 CYCLES PER SECOND. BLACK SPOT INDICATES THE APPROXIMATE BEGINNING OF A NEW CYCLE.
PLATE 10A  SCHLIEREN PHOTOGRAPH OF FLOW PATTERN ABOUT A CONE PROBE IN A TWO-DIMENSIONAL DIFFUSER. THE PROBE AT A DISTANCE OF 4 INCHES UPSTREAM OF THE THROAT. PERFORATED TO THROAT AREA = 4.27, EXIT TO THROAT AREA = 0.824
PLATE 10B  AS IN 10 EXCEPT THE PROBE AT A DISTANCE OF 2.5 INCHES UPSTREAM OF THE THROAT
PLATE 10C AS IN 10A EXCEPT THE PROBE AT A DISTANCE OF 1 INCH
UPSTREAM OF THE THROAT
PLATE 10D  AS IN 10A EXCEPT THE PROBE AT A DISTANCE OF 0.5 INCH UPSTREAM OF THE THROAT.
An Experimental Study of Perforated Intake Diffusers at Mach Number of 2.50

J.H. T. Wu, Sept. 1960 29 pages 10 plates 23 figures

1. Air Inlets, Supersonic 2. Nozzles, Supersonic, Perforated
3. Diffuser, Supersonic
I. Wu, J. H. T. II. UTIA Technical Report No. 69

A two-dimensional and an axisymmetric diffuser with perforations on the convergent portion were tested at a free-stream Mach number of 2.50. Schlieren photographs have shown that contrary to the simplifying assumption of a single normal shock, a prominent lambda shock appears in the convergent portion and multiple shocks occur in the divergent portion. High speed motion pictures have shown that both shock systems oscillate. No total pressure losses were detected between the intake lip and the multiple shocks downstream of the throat but the overall total pressure losses, were much higher than those based on the normal shock theory. The wall static pressures were the same as those measured along the diffuser axis providing the shock is swallowed and coincided with the pressure distribution of an isentropic flow except in the vicinity of the throat. A long throat did not reduce the total pressure losses. The most effective location of the perforations was found to be in a region just upstream of the throat.

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