Diffusion in a Supersonic Compressor Stator by Means of Aerodynamic Control

Vito D. Agosta*
Polytechnic Institute of Brooklyn, Farmingdale, N. Y.

Since at present the art of supersonic combustion has not been developed to an operational level, air discharging from a compressor rotor must be diffused to low subsonic speeds. Two problems are introduced, namely, that of starting a supersonic diffuser and the attainment of good performance at both design and off-design point operation. Application of aerodynamic control methods to the diffuser passage is investigated as a solution to these problems. A two-dimensional Mach 3 supersonic diffuser was designed and built which included bypass flow passages to satisfy starting requirements. The experimental investigation was conducted so that the maximum pressure recovery was achieved for minimum bypassed air. In addition, analysis indicates that a larger model would give an increase of mass flow through the diffuser without increasing losses in the pressure recovery. The maximum pressure recovery obtained for Mach 3 was 59% as compared to the theoretically calculated value of 62%, whereas the mass flow through the diffuser in the scaled-up model was a maximum of 85%.

Nomenclature

M = Mach number
A = area
p = pressure
R = pressure recovery

Subscripts

st = starting
t = total
th = throat
s = static

Introduction

THE potential advantages of employing impulse supersonic compressors in turbojet engines have created considerable interest for their use in high-speed vehicles. Some of the more significant advantages of this compressor design are the large rotor-tip-to-hub-diameter ratio, the high pressure ratio attainable per stage, and the use of supersonic inlet velocity. These features permit large quantities of air to be compressed in relatively small hardware with small inlet losses.

The operating characteristics of a compressor must be matched to those of the inlet. For a vehicle traveling at supersonic speeds and using a compressor operating with subsonic flow at its entrance, a variable geometry inlet should be employed in order to obtain good performance after starting and at off-design point operation. However, with supersonic flow at the entrance of the compressor, a constant geometry inlet with good performance at both design and off-design point operation can be considered. The compressor-inlet performance map would be quite different for both of these cases, the latter permitting a large variation in mass flow with relatively constant pressure recovery. In the impulse-type compressor, work is imparted by the rotor to the supersonic air flow with little or no static pressure rise. Since the art of supersonic combustion has not been developed to an operational state, then of necessity the supersonic flow from the compressor rotor must be diffused to low subsonic speeds.

Theoretical and experimental investigations[1–8] on supersonic compressors were made several years ago at the National Advisory Committee for Aeronautics. In an initial confidential report, Ferri discussed the attributes and the general problems of supersonic compressors and supersonic compressor design and made suggestions that were pursued in subsequent investigations.[9–11] It has been found experimentally that good performance is possible from impulse rotors alone[6,7]; however, when attempts were made to diffuse the supersonic flow, poor stage efficiencies resulted. The major problem lies, therefore, in determining the pertinent factors that influence good supersonic diffuser design. Since the flow conditions at the supersonic diffuser inlet are prescribed by the impulse rotor design, it then becomes important to consider the rotor in the diffuser design. Because of the large rotor-tip-to-hub-diameter ratios used, a nonhomogeneous supersonic flow field is generated in the rotor. Experimental investigations on the effect of nonhomogeneous fluid flow fields and wakes on supersonic diffusion and starting have been performed at the Polytechnic Institute of Brooklyn. In this report, only the theoretical and experimental investigation on the performance of a supersonic diffuser with aerodynamic control is presented. For simplicity, the flow was taken to be axial at the diffuser inlet; i.e., no swirl component was present.

Supersonic Diffuser Considerations

A supersonic stream can be decelerated into a subsonic one by means of a strong shock. The loss in pressure recovery inherent in such a process increases to prohibitive values as the Mach number increases. By decelerating the supersonic stream to a low Mach number before a shock occurs, the pressure recovery is thereby increased. In a convergent-divergent diffuser, the minimum area is limited by starting requirements. Therefore, for this type of diffuser passage, either a loss in the pressure recovery is accepted or some means of preventing choking during starting is found.

The methods employed in the past to prevent choking can be classified in two categories. One class of diffusers uses external supersonic diffusion and has been used mainly for inlets of jet engines, which employ single units that permit spillage. Where multiple units are required, e.g., in a stator cascade of a supersonic compressor, spillage may be prevented. A second class of diffusers uses internal supersonic

Presented as Preprint 64-246 at the 1st AIAA Annual Meeting, Washington, D. C., June 29–July 2, 1964; revision received March 12, 1965. This research was conducted under the sponsorship of the Office of Naval Research under Contract No. Nonr 839(21), Project No. NR094-215. The author wishes to acknowledge the suggestion of this investigation by Antonio Ferri, who also acted as principal investigator, and to thank M. Visich Jr. for the many helpful discussions and A. Martellucci and J. DeCarlo for their valuable assistance in the performance of the tests.

* Professor, Aerospace Engineering. Member AIAA.

diffusion in convergent-divergent passages. In order to prevent the large loss in pressure recovery concomitant of starting requirements, geometry of the convergent section of the passage is varied during starting.

Two additional approaches can be considered so that an aerodynamic change is produced in the contraction ratio of the main stream. In the first approach the throat of the passage is designed on the basis of starting considerations, and thus the Mach number therein, if no modification is introduced, is large. In order to decrease the Mach number at the throat, additional mass flow is introduced and passed through the minimum section. This secondary mass flow is added upstream of the throat so that the contraction ratio is increased. In the second approach considered, the throat of the main diffuser is designed for the condition of maximum efficiency after starting. A secondary throat (or bleed ports) is added to the converging passage and is used to relieve starting requirements. Therefore, during the starting process, air passes through the main and secondary throats, whereas after starting, by means of aerodynamic control, the air is forced to pass primarily through the main throat.

An evaluation of these two approaches is given below. Where secondary flow is used, in order to minimize losses, the stagnation pressure must be about equal to that of the main stream. In addition, the amount of secondary mass flow which must be added in order to change the contraction ratio is a function of the additional contraction required. For high Mach numbers, the secondary mass flow is of the same order of magnitude as that of the main stream and is therefore excessive. For these reasons the first approach does not appear very promising.

In the second approach the diffuser passage is designed for the maximum efficiency. During the starting process, the air is partly directed through the diffuser passage and partly spilled through the secondary throat or ports. For this case an aerodynamic mechanism must be devised so that, after starting, the hitherto spilled air is deflected into the diffuser passage. Let the convergent portion of a convergent-divergent diffuser be modified by perforations. During the starting operation, part of the subsonic flow behind the normal shock passes through the perforations, thereby relaxing minimum area requirements. As the shock is swallowed and supersonic flow established in the convergent portion of the diffuser, it is found that the spillage through the perforations decreases. Therefore, the perforations act as aerodynamic valves that permit starting by allowing spillage in the subsonic flow regime. Once the diffuser is started, the spillage is decreased in the supersonic flow regime.

Several past investigations[13–18] on perforated convergent-divergent supersonic diffusers have shown results that are quite encouraging; namely, high pressure recoveries and low spillage have been attained during operation. In addition, starting has been accomplished, and Mach numbers near one have been attained near the minimum section with good stability characteristics. Although the feasibility of using such a diffuser in multiple units, e.g., cascades in supersonic compressors, is mentioned by Evvard et al.,[14] there is no indication that such studies have been pursued.

A particular disadvantage of this type of diffuser is that spillage from the convergent portion of the diffuser occurs during normal operation. From the available experimental data, a graph was plotted of the peak pressure recovery vs the mass flow ratio through the diffuser (Fig. 1). Also plotted were calculated points that allow for losses incurred in a normal shock at the maximum contraction ratio for starting and 10% loss in the subsonic part of the diffuser. Mach number lines were drawn between the calculated and experimented points. Even though the data are quite meager, there is a trend indicated; namely, as the Mach number increases, both the peak pressure recovery and mass ratio through the diffuser decrease. In view of this situation, this device, used as an inlet and venting to the atmosphere, does not offer a sufficient advantage in pressure recovery to offset the loss in intake mass flow.[13]

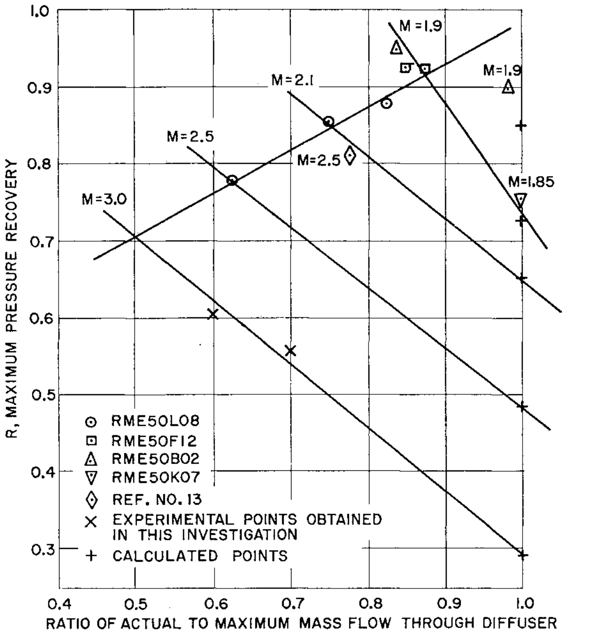

Fig. 1 Performance characteristics of supersonic diffusers with spillage.

By equalizing the pressure across the perforations, the spillage can be decreased once the diffuser has started. However, because of the existence of a pressure gradient in the convergent passage where the perforations are located, pressure equalization is difficult to acquire in the present diffuser designs. A method by which these difficulties can be overcome is described in the following section.

Design and Construction of the Supersonic Diffuser Test Model

In a supersonic compressor, the flow leaving the last-stage rotor is turned to the axial direction and diffused before entering the combustor. It has been shown in a previous study at the Polytechnic Institute of Brooklyn Aerodynamics Laboratory that supersonic turning, including three-dimensional effects, can be achieved without serious loss in pressure recovery.[10] In order to separate the problems of supersonic turning from those of diffusion, it is assumed that the sequence of events following the compressor rotor are turning, and then diffusion acting independently. The effects of centrifugal forces on the fluid flow field are neglected. In view of these assumptions, a two-dimensional approach is taken. The main effort lies, therefore, in investigating the problem of starting and subsequent diffusion in an internal passage.

The two-dimensional diffuser model is shown schematically in Fig. 2. Double wedges are used at the inlet to obtain the initial part of the internal supersonic diffusion. These are followed by air bleed passages that capture the spilled air during the starting operation. In order to increase the pressure aft the double wedges, air feedback passages fluid-dynamically coupled to the bleed passages are provided. The amount of feedback is controlled by means of a valve connecting the bleed and feedback plenum chambers. The main diffuser passage is centrally located between the bleed passages and behind the double wedges. It can be throttled independently of the other passages. The main-diffuser-passage walls correspond to the hub and shroud surfaces of a bladeless axially symmetric supersonic compressor channel.

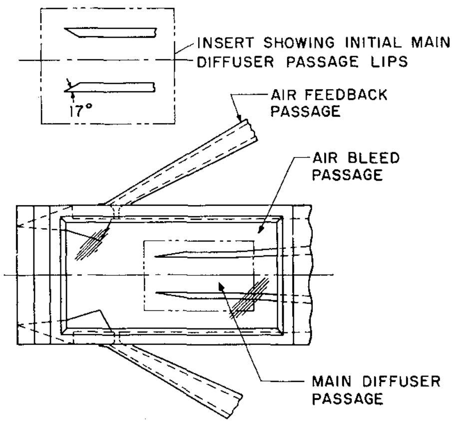

Fig. 2 Two-dimensional supersonic diffuser.

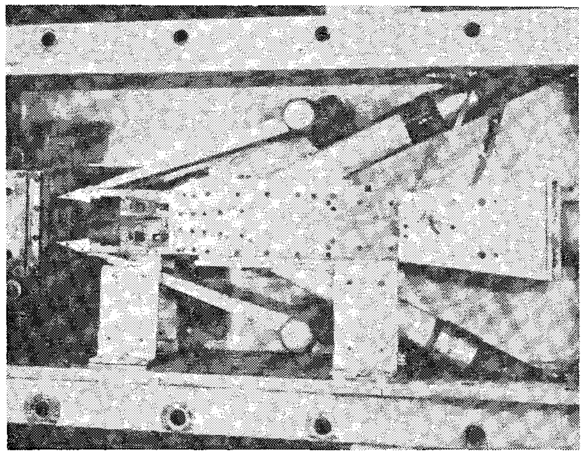

Fig. 4 Compressor model showing air bleed, feedback, and main diffuser passages.

The wakes generated from the blades upstream, comprising the turning passages, form radial viscous fluid elements in the bladeless concentric passage. Uninterrupted viscous diffusion of these wakes insures a flow becoming increasingly homogeneous, thereby allowing for efficient supersonic diffusion.[12]

The diffuser was designed for Mach 3 operation. The basic inlet shock configuration is shown in Fig. 3 and consists of two waves generated by the double wedge intersecting and merging into a single wave of equivalent strength. The single wave is intercepted by the main-diffuser-passage lip where, in the ideal case, it is cancelled. Under these ideal conditions, the static pressures are equal across the slip surface generated at the aft edge of the double wedge. A point of triple intersection is formed, i.e., the main-diffuser-passage lip, the oblique shock wave, and the slip surface. By throttling the main diffuser passage, a normal shock is located at the minimum section of the main diffuser passage. The main stream Mach number is decreased from 3.00 to 1.52, thereby decreasing the loss in pressure recovery due to normal shocks. The pressure recovery behind the normal shock in the minimum section of the main diffuser passage is estimated ideally (and considering no shock reflections) to be about 80%. Subsonic diffusion then continues to approximately $M = 0.3$, and the pressure recovery reduces to about 70%. Where a family of pseudoshocks occur in the main diffuser passage, a further decrease in the pressure recovery is realized. It has been shown by Conners[19] et al., in an investigation of flow at the throat of a two-dimensional diffuser at $M = 3.85$, that a local flow separation occurred with turning at the cowl lip. This condition was not relieved by use of a shock-cancellation surface downstream of the turn. Only by the use of the local application of wall suction and the relocation of the impinging shock downstream of the shock-cancellation surface was this condition relieved. It was also found by Conners that, for maximum total pressure recovery, a complex system of oblique shock waves existed instead of a single normal shock. It was initially attempted in this investigation to use a shock-cancellation surface and to study its behavior. The shock-cancellation surface is shown in the insert of Fig. 2. A discussion of the model operation using different surfaces at the main diffuser passage lip is given below.

In order to permit starting, the projected area of the main diffuser and the bleed passages normal to the axial flow direction was made sufficiently large to allow a normal shock at the model inlet to be swallowed. After starting is accomplished, supersonic flow occurs in the inlet of the model, and the described oblique shock-wave configuration is established. Since initially the pressure in the region behind the wedge and at the entrance of the bleed passage is different from that on the surface of the second wedge, a Prandtl-Meyer expansion occurs. The slip surface, therefore, does not intercept the main diffuser passage lips. By recirculating part of the bleed air through the feedback system, the pressure in the region increases, thereby raising the slip surface. Ideally, when the pressure in the region increases to a value equal to

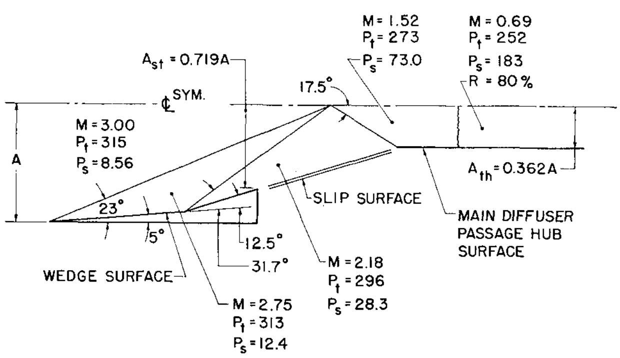

Fig. 3 Inlet shock configuration.

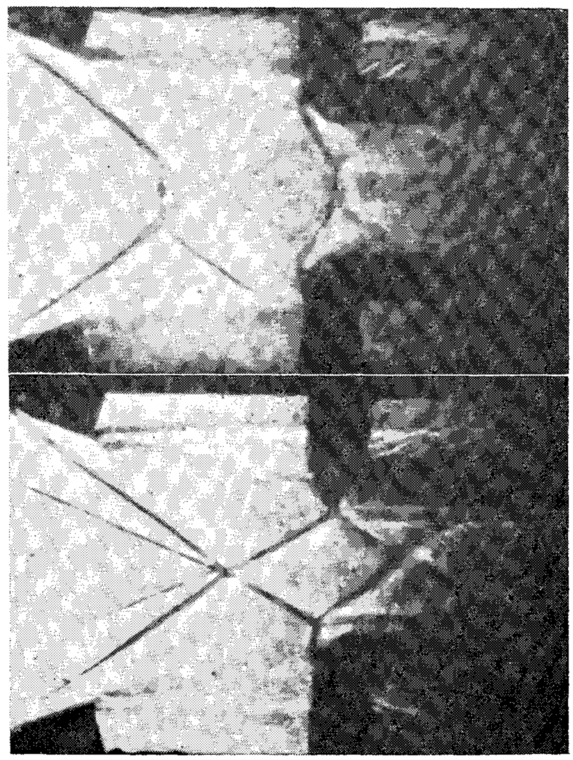

Fig. 5 Fluid flow field during starting.

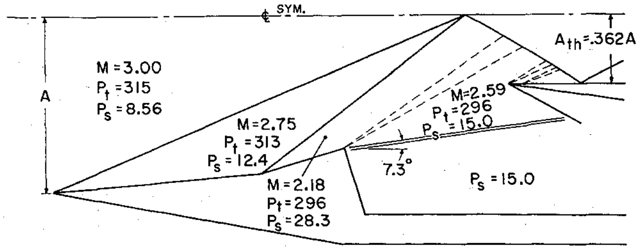

Fig. 6 Analysis of fluid flow field during starting.

that on the second wedge, the slip surface raises to form a continuous sheet from the second wedge to the triple intersection point. The minimum section of the main diffuser passage must, therefore, be large enough to pass all of the captured air after starting has been accomplished.

For the ideal case, after starting and pressure equalization is obtained across the slip surface, no air bleed occurs. However, because of viscous effects in the slip surface, some air bleed is desirable and can be attained by adjusting the valves in the bleed system. The model is shown in Fig. 4, and a physical description of the model is given below.

The model has a 2.90-in. span, 3.00-in. depth, and a chord of 26.00 in. from the entrance downstream to the instrumentation station. Glass plates are installed at the sides of the compression wedges to permit visual observation of the shock-wave configurations. The pressure instrumentation consists of static pressure taps installed along the lower contoured wall of the model and a total pressure rake and static taps at a station 26.00 in. downstream of the inlet. Two 2-in. pipes are used to connect the bleed passages to the bleed plenum chamber. The bleed plenum chamber is made of a 6-in. pipe 4 ft long. A 3-in. pipe is used to exhaust the plenum chamber to the atmosphere. A 3-in. gate valve and a $1\frac{1}{2}$-in. globe valve are used to adjust the amount of air bled from the model. One-and-one-half-inch piping is used for the feedback system. A feedback plenum chamber made of 3-in. pipe 3 ft long is connected to the bleed plenum chamber by a $1\frac{1}{2}$-in. pipe. A $1\frac{1}{2}$-in. globe valve is used to adjust the amount of feedback air. In addition, a supply of high-pressure air is connected to the feedback plenum chamber by a 1-in. pipe. A globe valve in the high-pressure line is used to meter the air to the feedback system. The air flowing through the main diffuser passage is measured by a calibrated venturi. In addition, the bleed air is measured by another flow meter.

Experimental Procedure and Preliminary Results

The model was tested in the Mach 3 wind tunnel, and data with the main diffuser and bleed passages wide open and feedback passages closed were obtained along with shadowgraphs of the inlet. The slip surfaces converged toward the center as predicted; however, a shock wave developed at the entrance to the main diffuser passage (Fig. 5). It was decided to analyze the fluid flow field in the inlet during starting. This was now possible since the pressure behind the entrance double wedges was available. Prandtl-Meyer expansion fans were generated from the inlet wedge corners, which interacted with the inlet oblique shock waves. The effects produced by these expansion fans were to curve the oblique shockwaves, to increase the downstream Mach number, and to cause the fluid flow field to diverge outward from the center. A sample calculation of the fluid flow field is shown in Fig. 6. It was found from this analysis that it was not possible for the oblique shock waves generated by the inlet wedges to sustain any reflection from the inclined surface of the main diffuser passage lips. Thus the main-diffuser-passage lip angles were modified so that the oblique shock waves would not detach.

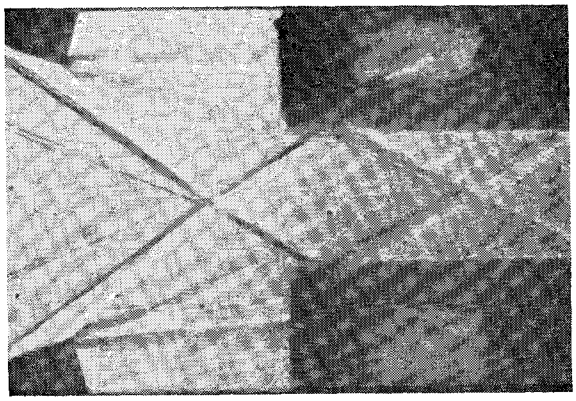

Fig. 8 Maximum bleed passage throttle setting.

The main diffuser and bleed passages were completely opened, and the feedback passage was closed. In Fig. 7 a shadowgraph of the fluid flow field in the inlet is shown. It is seen that the oblique shock waves are attached to the main-diffuser-passage lips as predicted. In this test about 60% of the flow spilled through the bleed passages, thereby permitting starting. In the next test the bleed passages were throttled and the valve setting for maximum throttling determined. It is seen in Fig. 8 that the slip surfaces tend to converge toward the center. The spillage was decreased to about 50% by this operation.

It was then decided to employ feedback to decrease spillage. The number of operations that had to be performed in order to start the model increased with each additional throttling or feedback device. It was necessary to optimize a starting procedure; therefore, several starting sequences were tried. It was felt that it would be advantageous to have feedback always turned on during the starting procedure. In this

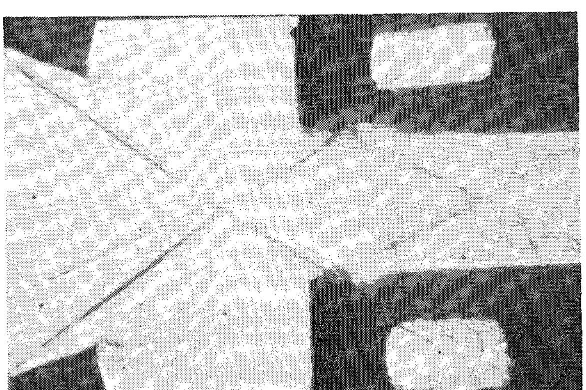

Fig. 7 Main diffuser and bleed passages open, feedback passage closed.

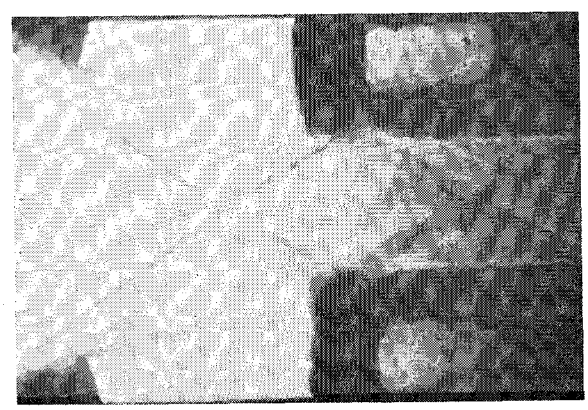

Fig. 9 Optimum bleed passage throttle setting, feedback passages open.

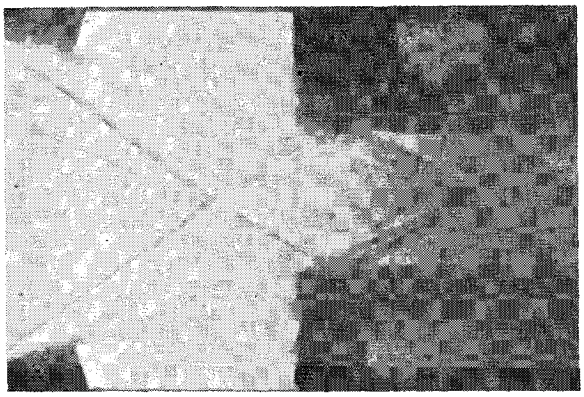

Fig. 10 Optimum bleed passage throttle setting, optimum main diffuser passage throttle setting, feedback passages open.

manner a physical operation would be eliminated. Moreover, an automatic compensating means would be imposed on the spillage; namely, since the feedback system was fluid-dynamically coupled to the bleed passages, then, as the spillage increased, the amount of feedback increased, which tended to decrease the amount of spillage. A fluid field configuration that was stable equilibrium was thereby attained in the inlet and was based on the pressure in the region behind the inlet wedges. Therefore, in the following tests the main diffuser, bleed, and feedback passage valves were completely opened before starting the wind tunnel. After starting, the bleed passage throttle setting was optimized. In Fig. 9 a shadowgraph of the inlet for the foregoing test is shown. It is seen that, by employing feedback, the slip surfaces converge significantly more than they did during the previous rest, as shown in Fig. 8. For both of these tests the oblique shock-wave configuration is essentially unaltered from that shown in Fig. 7 and dictated from design considerations.

The final test in this series included throttling the main diffuser passage. This operation came last in the starting procedure. A shadowgraph of the inlet during this test is shown in Fig. 10. The "normal" shock in the main diffuser passage is seen to be rather inclined; thus it is suspected that a family of pseudoshocks existed downstream in the passage. The spillage was 25% and the pressure recovery 46%.

It was assumed that a significant loss was due to viscous effects in the slip stream; thus the main diffuser passage was contracted. This modification tends to increase spillage; however, a pressure increase in the region aft the inlet wedges is generated which in turn tends to decrease spillage. The net effect is difficult to determine analytically.

Tests were run employing the same procedure as that used in the second series. In Fig. 11 a typical shadowgraph of the flow field is shown for this series of tests. The pressure recovery increased to 55% and the spillage to 31%.

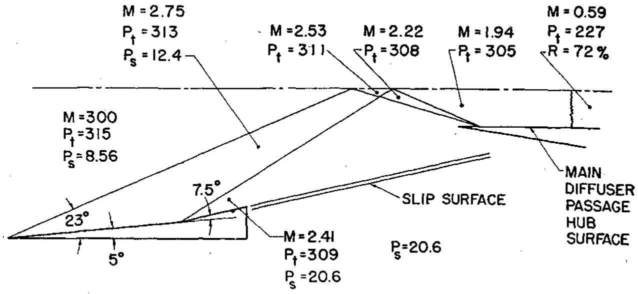

Fig. 12 Modified flow field.

Diffuser Modification and Analysis

It was felt that an increase in pressure recovery could be attained by modification of the diffuser to eliminate the expansion fan originating at the aft edge of the second wedge. Thus the angle was reduced from 12.5° to 7.5°, thereby reducing the pressure on the second wedge surface from 30 to 20 psia. In this manner the pressure in the bleed port area is approached and the expansion fan partially eliminated. In addition, shock-wave expansion-wave interaction is minimized, thus generating a more uniform field behind the oblique shock waves.

Since the direction of the slip surface is changed by the preceding modification, the main diffuser inlet lips are relocated in order that the same amount of air is bled. The main diffuser inlet lips are moved downstream to a position where the new oblique shock waves intersect the main diffuser passage lips. In addition, by moving the main diffuser inlet along a streamline, the spillage remains essentially constant assuming that the flow field does not vary significantly. It is recognized that all of these changes alter the flow field. However, since the effects of all of these changes coupled together cannot be calculated theoretically, the effects caused therefrom were calculated independently, and then all of the results were considered as a whole. The flow field is shown schematically in Fig. 12.

The two-dimensional flow analysis corresponding to that shown in Fig. 12 predicts a 72% pressure recovery behind the normal shock located at the throat of the main diffuser passage. However, it was not found possible to position a normal shock wave at the throat; a family of pseudoshocks always appeared. Since the compression in the subsonic part of the diffuser is not isentropic, one can assume that 10% of the total pressure is lost.[20] Thus, the final theoretical pressure recovery that can be achieved is 62%.

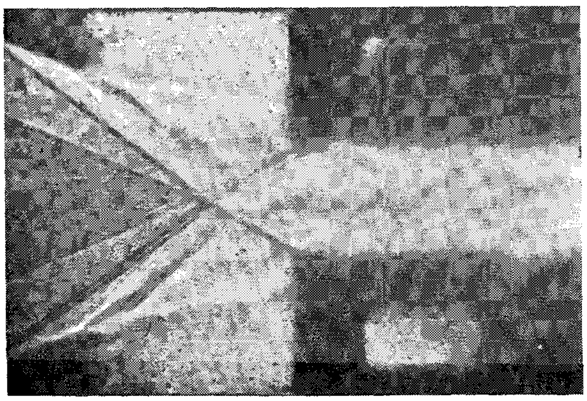

Fig. 11. Fluid flow field with optimum throttle settings for contracted main diffuser passage.

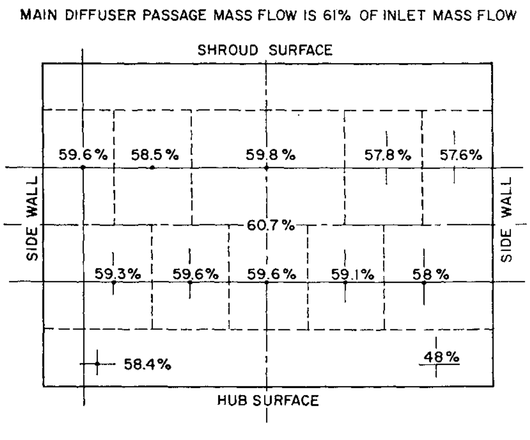

Fig. 13 Pressure recovery distribution at exit of subsonic part of main diffuser passage.

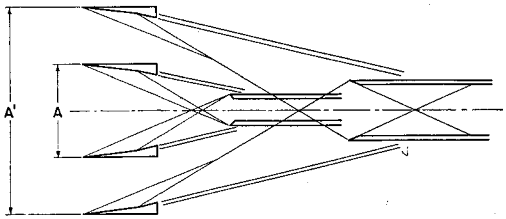

Fig. 14 Superposition of two diffuser arrangements to demonstrate scaling.

Results

The Mach number at the instrumentation station was 0.30. In Fig. 13 the local variation in pressure recovery across the station is shown. It is seen that the pressure recovery was reasonably uniform, the average value being 59% as compared to the theoretical value of 62%. The difference in these values may be due to the inability to position a normal shock at the diffuser throat section. The mass flow through the diffuser was 61%.

The results obtained suggest that approximately the same pressure recovery could be obtained but with a lower percentage bleed flow if the area ratios were changed. By enlarging the capture and main diffuser passage areas, it is possible to increase the percent mass captured in the main diffuser passage (Fig. 14). In addition, by maintaining the fluid fields geometrically similar, and bleeding the viscous wake due to the slipstream aft the wedges, the losses due to viscous mixing remain constant. This is accomplished by positioning the main diffuser passage lip on a given streamline that lies inboard of the wake. It is seen that the initial oblique shock waves fall further downstream in the main diffuser passage; however, no major change occurs in the total pressure recovery of the diffuser. By this artifice the mass recovery increases to 85%.

Conclusions

The results from this investigation indicate that the system of air bleed ports in the supersonic compressor diffuser to relieve starting conditions is operative. Aerodynamic control of the diffuser is achieved by varying the bleed port pressure.

References

[1] Kantrowitz, A., "The supersonic axial-flow compressor," NACA Rept. 974 (1950).

[2] Ferri, A., "Preliminary analysis of axial-flow compressors having supersonic velocity at the entrance of the stator," NACA RM L9G06 (1949).

[3] Wright, L. C. and Klapproth, J. F., "Performance of supersonic axial-flow compressors based on one-dimensional analysis," NACA RM E8L10 (1949).

[4] Liccini, L. L., "Analytical and experimental investigation of 90° supersonic turning passages for supersonic compressors or turbines," NACA RM L9G07 (1949).

[5] Goldstein, A. W., "Axisymmetric supersonic flow in rotating impellers," NACA Rept. 1083 (1952).

[6] Tysl, E. R., Klapproth, J. F., and Hartmann, M. J., "Investigation of a supersonic-compressor rotor with turning to axial direction. I—Rotor design and performance," NACA RM E53F23 (1953).

[7] Goldstein, A. W. and Schacht, R. L., "Performance of a swept leading edge rotor of the supersonic type with mixed flow," NACA RM E52K03 (1953).

[8] Sterrett, J. R., "Experimental investigation of an impulse-type supersonic compressor rotor having a turning of 73° at the mean radius," NACA TN 4252 (1958).

[9] Vaglio-Laurin, R., "Three-dimensional effects on supersonic compressors. Part 2—The analytical method and its application for rotating and fixed coordinate systems," Polytechnic Institute of Brooklyn, PIBAL Rept. 234 (October 1953).

[10] La Rocca, A., "Three-dimensional effects in supersonic compressors. Part 3—The design of two stator passages and comparison between theory and experiment," Polytechnic Institute of Brooklyn, PIBAL Rept. 271 (December 1954).

[11] Buongiorno, C. and Visich, M., Jr., "Some requirement for obtaining efficient diffusion in supersonic compressors," Polytechnic Institute of Brooklyn, PIBAL Rept. 315 (December 1956).

[12] Agosta, V. D. and Martellucci, A., "An investigation on diffusion in a supersonic compressor stator passage in the presence of rotor blade wake and hub and shroud boundary layer," Polytechnic Institute of Brooklyn, PIBAL Rept. 394 (October 1957).

[13] Brown, D., "An experimental investigation of the pressure recovery of a perforated-wall internal-compression intake at the free stream Mach number of 2.5," Univ. of Toronto TN 18 (May 1957).

[14] Evvard, J. C. and Blakey, J. W., "The use of perforated inlets for efficient supersonic diffusion," NACA TN 3767 (1956).

[15] Wyatt, De M. D. and Hunczak, H. R., "An investigation of convergent-divergent diffusers at Mach number 1.85," NACA RM E50K07 (1951).

[16] Hunczak, H. R. and Kremzier, E. J., "Characteristics of perforated diffusers at free-stream Mach number 1.90," NACA RM E50B02 (1950).

[17] Weinstein, M. I., "Investigation of perforated convergent-divergent diffusers with initial boundary layer," NACA RM E50F12 (1950).

[18] Baughman, L. E. and Gould, L. I., "Investigation of three types of supersonic diffuser over a range of Mach numbers from 1.75 to 2.74," NACA RM E50L08 (1951).

[19] Conners, J. F. and Woollett, R. R., "Some observations of flow at the throat of a two-dimensional diffuser at a Mach number of 3.85," NACA RM E52104 (1952).

[20] Oswatitsch, K I., "Pressure recovery for missiles with reaction propulsion at high supersonic speeds (the efficiency of shock diffusers)," NACA TM 1140 (1947).