N.A.E.

R. & M. No. 2827
(11,868)
(A.R.C. Technical Report)

MINISTRY OF SUPPLY

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G. H. Lean, B.Sc., D.I.C.,
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LONDON: HER MAJESTY'S STATIONERY OFFICE
1954
PRICE 4*s* 6*d* NET

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Reports and Memoranda No. 2827[*]
March, 1945

Summary.—The present work continued that reported in Ref. 1 and extended some of the results there described to lower entry Mach numbers (1·3 to 1·9). It was found, as in Ref. 1, that with a parallel entry duct followed by a straight divergent diffuser of 10 deg total angle the flow inside the parallel tube was supersonic provided the outlet pressure of the diffuser was less than a certain critical value (about 0·93 of the upstream pitot pressure). In this case the mass flow of air through the tube was equal to that calculated, assuming that all the air incident on the internal section of the tube entry passed through it. For pressures higher than the critical value the flow became subsonic at the duct entry, a shock-wave was formed at the entrance lip and the rate of airflow through the tube decreased.

Similar results were obtained for a uniformly divergent tube of 7 deg total angle; in this case, however, an outlet pressure equal to 0·97 of the upstream pitot pressure was attained before the shock-wave left the lip.

For outlet pressures less than the critical the flow was supersonic for a distance inside the duct entry depending on the outlet pressure, the flow becoming subsonic further along the duct. The assumption of unidimensional flow in the duct led to results which showed considerable disagreement with the observed pressure distribution more especially in the subsonic flow region. This could be explained by assuming that the flow in this region separated from the duct wall.

The results of tests on models of two forms of annular entry (the Q1 and E24/43 entries respectively) showed that two types of flow régime were possible depending on the outlet pressure. For low outlet pressures a shock-wave was formed at the lip of the entry and the flow passed into the entry but for higher outlet diffuser pressures there was no distinct shock-wave from the annular lip and the flow through the annulus was reversed. The outlet diffuser pressure at which the flow changed direction for each entry tested was about 0·5 of the free-stream pitot pressure and was thus considerably lower than for the unobstructed type of duct.

With the airflow passing into the annulus the flow through the Q1 entry was independent of the outlet pressure over the range of Mach numbers tested and was about 0·88 of the flow through an area equal to the intake area in the free stream. For the E24/43 entry the airflow decreased as the outlet diffuser pressure increased probably due to changes in the boundary-layer thickness at the annulus.

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1. *Introduction.*—A simple type of propulsive power unit had been suggested to enable aircraft or projectiles to travel at supersonic speeds. The power unit was essentially of tubular form, air flowing through the tube from an intake facing upstream and being compressed by a suitably shaped diffuser. At the outlet of the diffuser the temperature of the air was raised by injecting fuel into the air stream and the products of combustion were passed through a converging or diverging exit jet, the increased momentum of the issuing jet providing a forward thrust. Making

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certain assumptions with regard to the intake diffuser efficiencies, the total-head losses in the combustion chamber and the conditions at the exit of the jet, the thrust and efficiency for a unit of this type could be calculated for various operating speeds[2]. The present work was initiated mainly to examine the flow conditions near the duct entry of such a unit and to measure the diffuser efficiencies which might be expected in practice.

The diffuser efficiency has been defined as the ratio of the work done assuming adiabatic compression of the air between the inlet and outlet pressures to the loss in kinetic energy of the air passing through the diffuser. Thus if

p_1 is the static pressure of the air approaching the diffuser
p^* is the total head of the air at the diffuser exit
M_1 is the Mach number of the air approaching the diffuser,

the diffuser efficiency as defined above may be expressed as

$$\eta = \left\{ \left(\frac{p^*}{p_1} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right\} \Big/ \frac{\gamma - 1}{2} M_1^2 . \quad \cdots \quad (1)$$

The efficiency of a diffuser at a given Mach number therefore depends only on the ratio of the total head at exit to the diffuser to the upstream static pressure.

If the compression through the diffuser is assumed to take place by compression through a perpendicular shock-wave and subsequent compression of the air emerging from the shock-wave, it is possible to calculate p^*/p_1 provided the efficiency of the compression after the shock-wave is known. The efficiency of compression after the shock-wave (referred to as the 'subsonic diffuser efficiency') is defined in the same manner as for Equation (1), *viz.*:—

$$\eta' = \left\{ \left(\frac{p^*}{p_2} \right)^{\frac{\gamma-1}{\gamma}} \quad 1 \right\} \Big/ \frac{\gamma - 1}{2} M_2^2 . \quad \cdots \quad (2)$$

where p_2 is the static pressure after the shock-wave and

M_2 is the Mach number immediately downstream of the shock-wave.

The values of η assuming values of η' corresponding to 0, 40, 60, 80 and 100 per cent have been plotted in Fig. 1 for a range of Mach numbers. For $\eta' = 100$ per cent the compression behind the shock-wave is completely adiabatic and the values of p^* then correspond to the pitot pressure of the air in the free stream. The ratio of p^* to the pitot pressure for other values of η' are plotted in Fig. 4.

2. *Tests on Unobstructed Intakes.*—2.1. *Description of Wind Tunnel Used for Tests.*—The liners of a $2\frac{1}{2}$-in. square supersonic wind tunnel were modified so that Mach numbers from 1·3 to 1·9 could be obtained in the working-section. The side walls of the tunnel consisted of two stiff steel springs which were rigidly held at their downstream ends and inclined slightly to each other. They were each about 8-in. long and curved at their upstream ends where they were faired to shaped wooden blocks which formed the tunnel entry; a screw adjustment was provided so that the upstream ends of the springs could be moved towards each other and thus the throat section and the Mach number in the tunnel working-section changed. Whilst the form taken up by the springs was not the theoretical optimum, it appeared from pitot and static-pressure explorations that the conditions in the working-section were sufficiently uniform for the present tests.

2.2. *Test Arrangement.*—In the preliminary tests the entry took the form of a $\frac{3}{4}$-in. diameter glass tube which was mounted on the axis of the working-section. The downstream end of the tube was joined to a straight tapered divergent diffuser of 10 deg total angle and 1·5-in. exit diameter. The air passing through the tube and diffuser discharged into a parallel tube containing a measuring orifice. The end of this tube delivered to atmosphere and by blocking the exit or by connecting it to an exhauster, the pressure at the outlet of the diffuser could be varied.

The glass tube was chamfered on its inlet end at an angle of about 20 deg so that the flow inside the tube could be observed up to a short distance from the lip.

2.3. *Description of Flow at Intake.*—For outlet pressures small compared with the free-stream total head the flow in the parallel tube was supersonic, inclined shock-waves being formed in the tube (Photographs 1 and 2). As the outlet pressure was increased a faint shadow (which was too nebulous to be photographed) moved up the tube, the shock-waves disappearing as the shadow reached them until when it arrived at the upstream end of the tube a plane perpendicular shock-wave, which oscillated slightly, was formed across the lip. Further increase of pressure caused the shock-wave to move upstream of the lip and assume a marked curvature. (The flow photographs obtained with the shock-wave in this position were very similar to Photographs 4 to 6 of Ref. 1). Reduction of the inlet pressure caused the same series of changes described above to recur but in the reverse order.

It will be observed from Photographs 1 and 2 that at the junction of the lip shock-waves a Mach shock-wave was formed perpendicular to the axis of the tube. Also after each test there were well-marked dust deposits in the form of rings on the walls of the tube at varying distances from the front end. Comparing the positions of these rings with the shock-wave system it appeared likely that they corresponded with the points of reflection of the waves; the mean distance between the rings increased along the tube indicating that the shock-waves became gradually weaker.

3. *Results with a Parallel Entry.*—A series of flow measurements were made with the diffuser described above in position but with the glass tube replaced by a metal parallel tube of similar dimensions having its upstream end bevelled at an angle of 6 deg. (This angle was less than the critical wedge angle for which a head shock-wave would become detached for the present Mach numbers.) The flow through the tube was varied by blocking the exit of the diffuser and the total-head recovery at the diffuser outlet measured for five entry Mach numbers (Fig. 2).

With the tube in the supersonic condition the flow of air was equal to the calculated flow striking the internal section of the tube. When the shock-wave left the lip the mass flow decreased, and when the flow was reduced below a certain value depending on the Mach number, the flow in the tunnel itself became subsonic. (The curves of Fig. 2 were discontinued at these values.)

The ratio of the total-head recovery at the diffuser outlet to the upstream pitot pressure when a plane shock-wave was formed across the entry of the tube was plotted against the upstream Mach number (Fig. 4). Comparing the experimental curve with the calculated values it appeared that the subsonic efficiency of this form of diffuser was about 75 per cent and was approximately constant over the range of Mach numbers tested.

4. *Results with a Straight Tapered Entry.*—The parallel tube and divergent diffuser was replaced by a uniformly tapered entry having an upstream diameter of 0·5-in. and internal total angle of approximately 7 deg, the outside lip of the entry being inclined at 5·6 deg to the axis. As before the total-head recoveries at the diffuser outlet and the flows through the diffuser were measured at three entry Mach numbers (Fig. 3). Similar changes in the flow characteristics occurred to those already described with the parallel entry, the flow of air through the diffuser decreasing as soon as the shock-wave was formed at the lip. The total-head recoveries with the wave in this position gave a subsonic diffuser efficiency of about 90 per cent (Fig. 4). This value was higher than with the parallel entry probably due to reduced frictional losses with the smaller angle of the diffusing cone and the absence of losses associated with the sudden change of section at the end of the parallel tube.

5. *The Pressure Distribution inside the Intake.*—In order to examine the flow inside the tapered tube two thin plane glass windows were cemented into the sides, the windows being as flush as possible with the inside. To trace the flow condition inside the tube, a small disc 0·08-in. diameter was mounted on the axis of the duct perpendicular to the flow. With a Mach number of 1·45 and an outlet diffuser pressure equal to about 1·6 times the upstream static pressure a shock-wave was visible in front of the disc at 1 in. inside the entry (Photograph 3). With the disc at 1·5 in. inside the entry the shock-wave was absent (Photograph 4). Evidently the flow in the duct became subsonic between these positions.

Pitot and static-pressure measurements were made along the axis of the duct with various outlet pressures (Figs. 5 and 6). These indicated that as the outlet pressure was decreased, the length of the supersonic flow region increased, the pitot and static pressures being constant in this length and independent of the outlet pressure.

The calculated pressure distributions assuming uni-dimensional flow for an upstream Mach number of 1·5 were plotted for comparison in Figs. 5 and 6. These curves assumed a plane perpendicular shock-wave at some point in the duct, frictionless adiabatic expansion from the entry to the shock and a subsonic diffuser efficiency of 90 per cent after the shock-wave. The shock-wave was assumed to occur at the point at which the experimental pitot and static measurements began to differ from their values with lower outlet pressures. The variation of this position with the outlet pressure was compared with the calculated outlet pressures assuming varying positions of the perpendicular shock-wave (Fig. 7).

It was evident (Figs. 5, 6 and 7) that whilst the general trend of the experimental curves was similar to that obtained assuming unidimensional flow the disagreement (except for the case of the shock-wave at the lip where the agreement was moderately good) was too great to render such a simple description of the motion adequate. Evidently the inclined shock-waves associated with pronounced fluctuations of the pitot and static-pressure curves could not be neglected. Also the Mach numbers in the subsonic region along the axis of the duct were considerably higher than those calculated, indicating that the flow in this portion of the duct had separated from the wall due to the influence of the shock-waves on the boundary layer.

6. *Tests on Annular Type Intake Ducts.*—Models of two forms of annular entry, referred to as the Q1 and E24/43 (Fig. 8) entries respectively, were tested. Each intake consisted of a divergent tube and a central spindle carrying the head of the model, the gap between the head of the model and the lip of the tube serving as the annular intake. The divergent tube was connected to the $1\frac{1}{2}$-in. diameter parallel tube as for the previous tests. The most extensive series of tests was performed on the Q1 shaped entry at a Mach number of 1·88: these are described in detail below.

(a) *The Flow Instability.*—For low outlet pressures, the flow régime corresponded with that shown in Photograph 5, a shock-wave being formed at the lip of the annular entry as well as at the head of the model. This shock-wave system remained unchanged as the pressure was increased until at a critical value of the pressure the flow régime changed suddenly to that shown in Photograph 6. This change was accompanied by considerable pressure fluctuations and the direction of the airflow through the duct was reversed, air passing into the tunnel via the annulus.

When this occurred (Photograph 6) the well-defined shock-wave from the annulus disappeared and the head-wave showed a pronounced change of inclination at a short distance from the nose. When the outlet pressure was further increased this change of inclination moved towards the nose and the faint shadows behind this position became stronger (*cf.* Photograph 9). It was clear that the air issuing from the annulus formed a compression-wave which interacted with the head-wave and moved forward as the reverse flow became stronger.

With the flow régime corresponding to Photograph 6 established it was possible to reduce the outlet pressure until the reversed airflow through the annulus was zero (Photograph 7); further reduction of the outlet pressure invariably resulted in the return of the flow pattern corresponding to photograph 5.

(b) *Rate of Flow Measurements.*—The flow of air through the annulus and the outlet pressures at the downstream end of the divergent tube were measured (Fig. 9). With the flow passing into the annulus the flow rate was constant, whilst with the flow reversed the flow rate increased with the outlet pressure. The pressure fluctuations occurring during the transitional state made flow measurements difficult: on this account and because it was considered that such measurements had little practical value, no measurements were made in this condition.

For the Q1 entry the maximum outlet pressures increased with Mach number and the mass flow through the annulus remained fairly constant being equal to about 0·88 of the flow through an area in the free stream equal to the annulus. The E24/43 entry gave similar results (Photographs 11 and 12) but the maximum pressure recoveries were lower and the flow rates decreased as the pressure increased due probably to air escaping through the boundary layer.

(c) *Pitot Pressures Near the Annulus.*—The pitot pressure near the annulus were measured for various outlet pressures (Fig. 10). With the flow passing into the annulus the pitot pressure outside was greater than the free-stream pitot pressure and was independent of the outlet pressure. Inside the annulus, as the outlet pressure was raised, the pitot pressure decreased and showed pronounced fluctuations at points well inside. With the flow reversed the pitot pressures upstream of the annulus fell abruptly. (The indicated pressure then corresponded more nearly to the static pressure in this region.)

7. *Effect of Varying the Head Position.*—Because of the higher pitot pressure upstream of the annulus it was thought that greater pressure recoveries might result if the head of the model was moved downstream. This was found to be the case but at the maximum outlet pressure the flow became unstable oscillating between the flow patterns already described (Photographs 5 and 6). In Photograph 8 the flow was in this unstable state and both patterns are seen together. The frequency of oscillation varied with the amount of choking, the flow pattern corresponding to Photograph 6 persisting for a longer time as the choking became larger.

The pressure recoveries and flow rates were measured for the head in a number of positions relative to the duct (Fig. 11). It was found that the maximum pressure recovery occurred with the nose of the model at 0·38 in. from the annulus (*cf.* Photograph 10) and was then approximately equal to the free-stream pitot pressure. For positions further downstream the flow characteristics were similar to those occurring for the unobstructed duct already described.

The mass flow of air through the annulus with the head 0·5 in. inside the duct was less than that calculated for an unobstructed duct and a plane shock-wave was formed about 0·05 in. upstream of the lip. This was because the minimum cross-section of the annular duct was less than the minimum cross-section for sonic velocity.

8. *Discussion.*—For comparison with the present tests, the maximum pressure recoveries obtained with supersonic wind-tunnel diffusers of 5 deg total angle[3] and also the results for the present $2\frac{1}{2}$-in. tunnel diffuser liners were plotted in Fig. 4. These were considerably below the recoveries obtained with the ducts, due to the fact that in the tunnel the transition from supersonic flow in the diffuser did not take place in a perpendicular shock-wave but one in which the wave was considerably distorted near the walls.

The pressure recovery with a parallel-entry type duct followed by a 10 deg total angle diffuser for an upstream Mach number of 2·6 gave a considerably lower subsonic diffuser efficiency. The reason for this may have been the proportionately higher losses in the tube owing to slight asymmetry of the incident flow.

In an extract from ' Progress Report on Westinghouse Jet-Propulsion Motor ' dated 5th January, 1943, the subsonic diffuser efficiencies for a uniformly tapered diffuser of approximately 7 deg total angle over a range of Mach numbers between 0·5 and 1·0 was given as 84 per cent, and assuming this value the pressure ratio across the plane shock-wave at the lip at various supersonic Mach numbers was compared with the theoretical values. The value of 90 per cent for the subsonic diffuser efficiencies for the present tests was somewhat higher than this value: however, the difference in the ratio of the pressure recovery to the upstream pitot pressure for these two values was only about 2 per cent.

The tests on the annular type intakes were made at an approximate Reynolds number of 6×10^5 (based on the distance from the nose of the model to the annular lip). Assuming incompressible flow the calculated ratios of the boundary-layer thickness to the annular gap width in the models were about 1·7 times that in the full-scale, so that in this respect the model tests were not representative.

9. *Conclusion.*—The tests on the straight tapered divergent intake of 7 deg total internal angle indicated that when the conditions were such that a plane shock-wave was formed at the entry lip the pressure recoveries were about 97 per cent of the free-stream pitot pressures for Mach numbers between 1·3 and 1·9 and in this condition the mass flow through the tube was equal to that striking the lip. For pressures at the outlet of the intake greater than this the shock-wave was detached and the flow decreased. For lower pressures both the pressure explorations along the intake and the photographic evidence suggested that the flow was supersonic inside the tube and inclined shock-waves occurred in the flow. In this condition the assumption of unidimensional flow was inadequate, and it was probable that the collision of the shock-waves with the boundary layer produced an adverse pressure gradient which caused a considerable increase in the boundary-layer thickness.

The tests on the annular type intakes gave pressure recoveries considerably below those for the unobstructed type. Greater recoveries could be attained by reducing the distance from the nose to the annular lip but in this case the range of vision of the pilot, if accommodated in the head of the machine, would be seriously restricted. When the pressure was raised above a certain critical value the flow through the annulus was reversed and for positive mass flows less than the maximum, the flow near the annulus oscillated between two stable flow patterns and was accompanied by pronounced pressure pulsations.

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2	A. R. Howell and Marjorie Mettam	Note on the Performance of Propulsive Ducts. R.A.E. Technical Note Eng. 256.
3	.. ..	Report on General Construction and Results of Preliminary Trials on a Closed Circuit Variable Density Supersonic Wind Tunnel. N.P.L. Eng. Dept. Reference (E) T.P. 241. February, 1943.

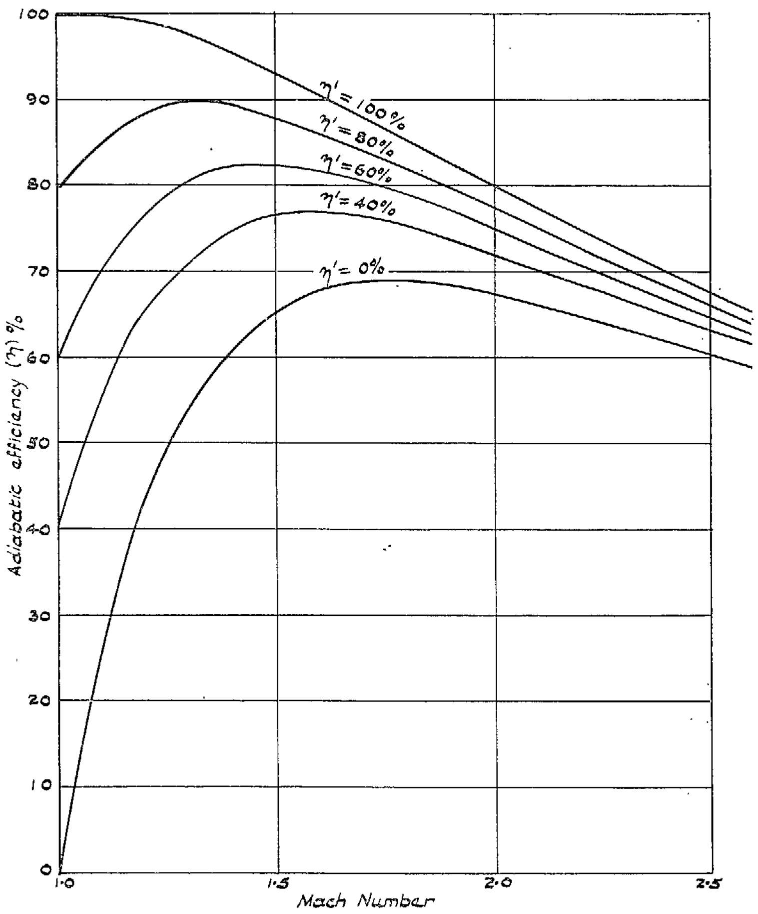


FIG. 1. Variation of adiabatic efficiency of a diffuser with Mach number assuming a plane shock-wave at the head of the diffuser and different subsonic efficiences.

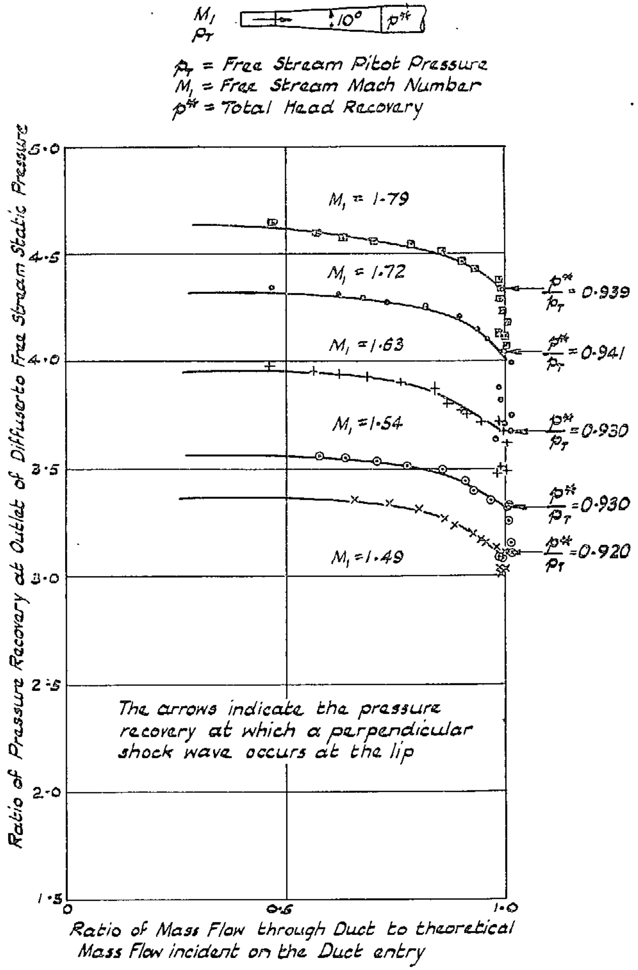


FIG. 2. Variation of total-head recovery with mass flow of air through a parallel tube followed by a 10-deg total angle uniformly divergent diffuser.

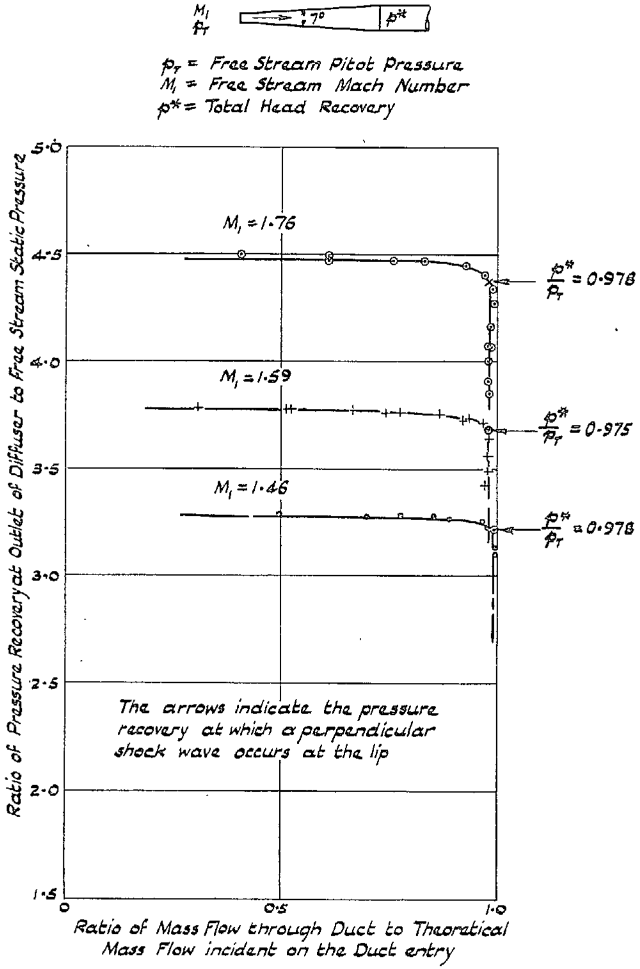


FIG. 3. Variation ot total-head recovery with mass flow of air through the 7-deg total angle uniformly divergent diffuser.

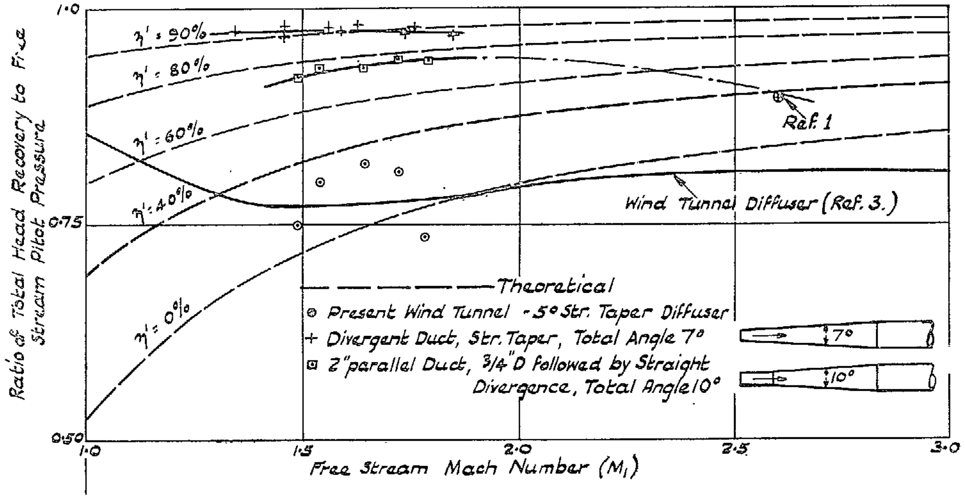
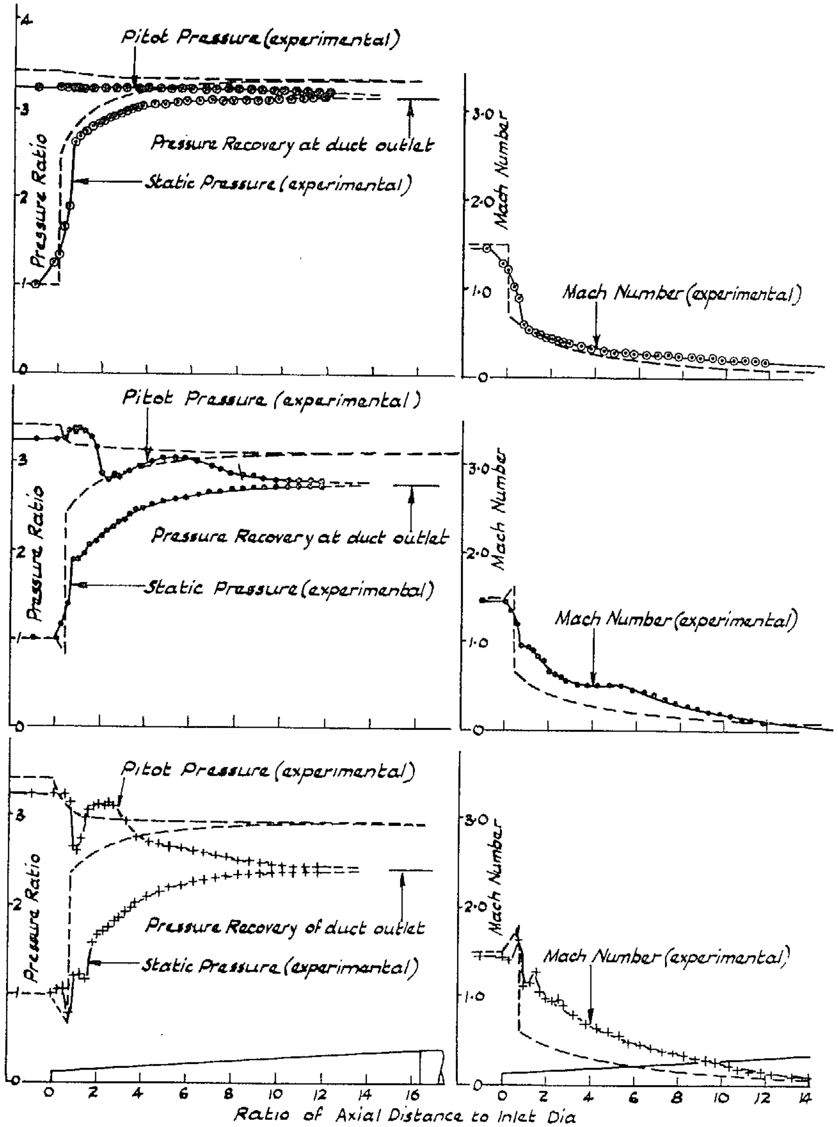
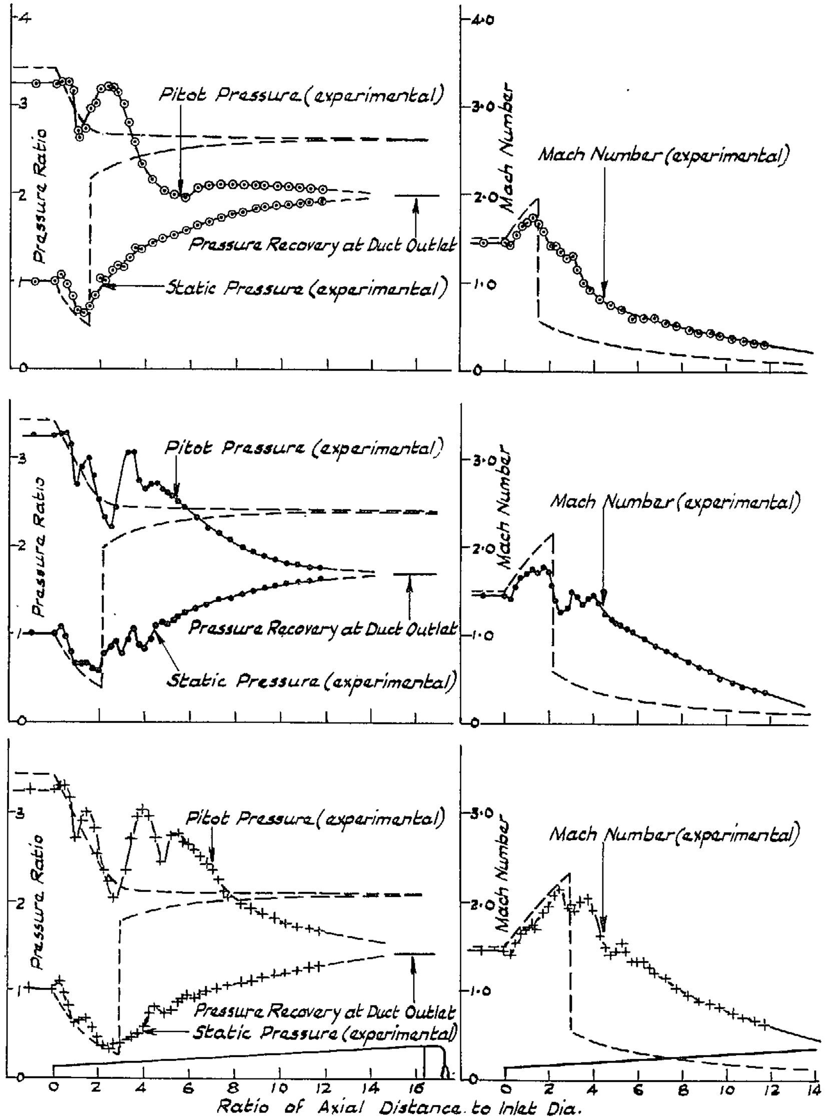


FIG. 4. Ratio of total-head recovery to free-stream pitot pressure assuming various subsonic diffuser efficiencies and a perpendicular shock-wave at the lip.



The dotted curves are the theoretical values assuming an upstream Mach Number of 1·5 (η' = 90%) and the occurrence of a plane shock wave at the positions indicated by the vertical lines

FIG. 5. Pressure and Mach number variation along axis of 7-deg uniformly tapered duct ($M = 1 \cdot 45$) for various outlet pressures.



The dotted curves are the theoretical values assuming an upstream Mach Number of 1·5 (η' = 90%) and the occurrence of a plane shock wave at the positions indicated by the vertical lines.

FIG. 6. Pressure and Mach number variation along axis of 7-deg uniformly tapered duct ($M = 1 \cdot 45$) for various duct pressures.

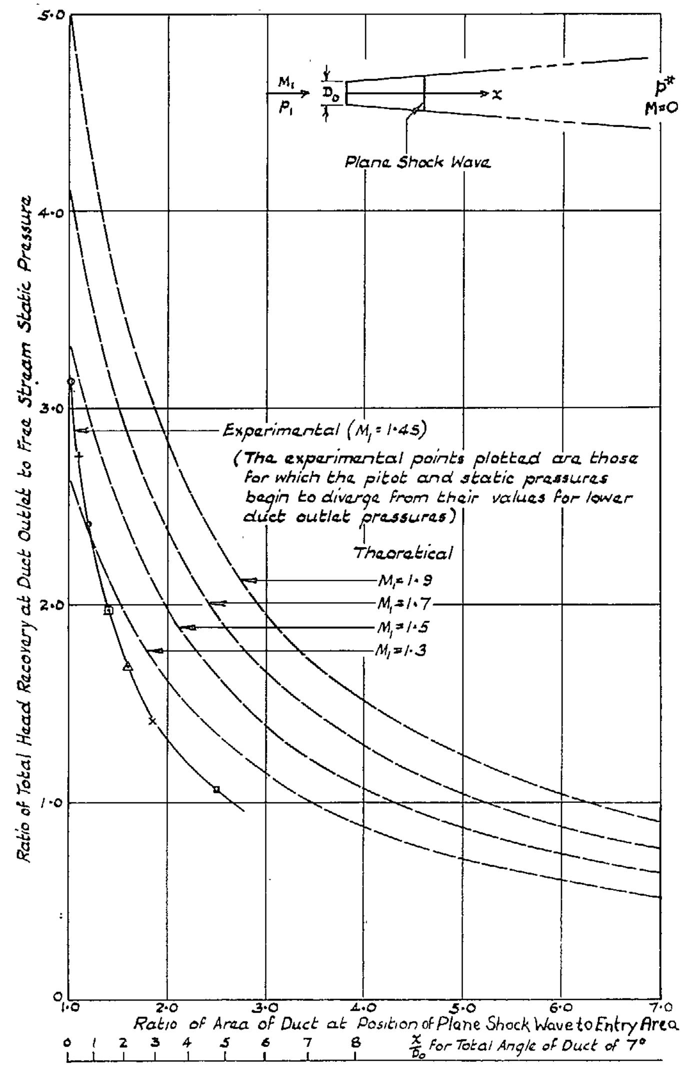


FIG. 7. Total-head recovery for perpendicular shock-wave at various positions in a uniformly divergent duct assuming 90 per cent subsonic diffuser efficiency.

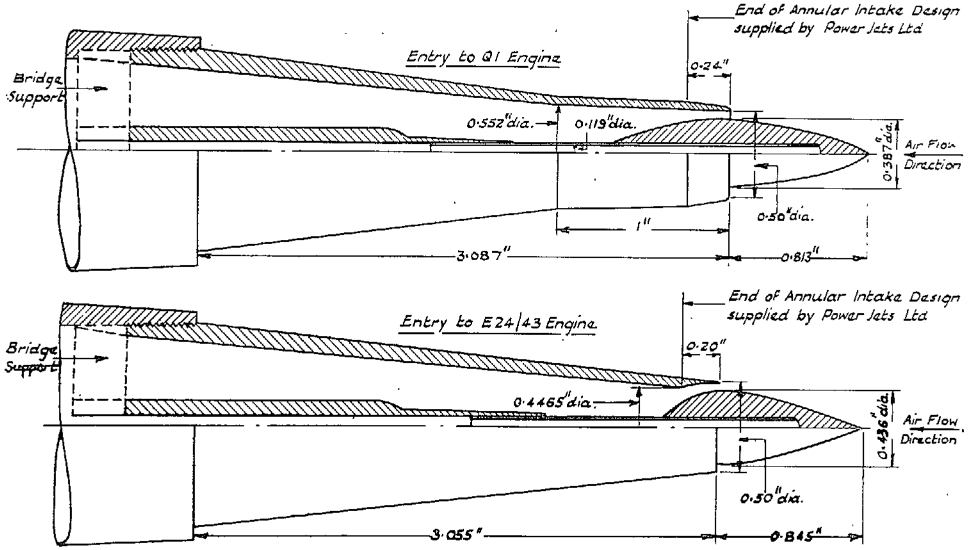


FIG. 8. Sketch of two shapes of annular entry tested.

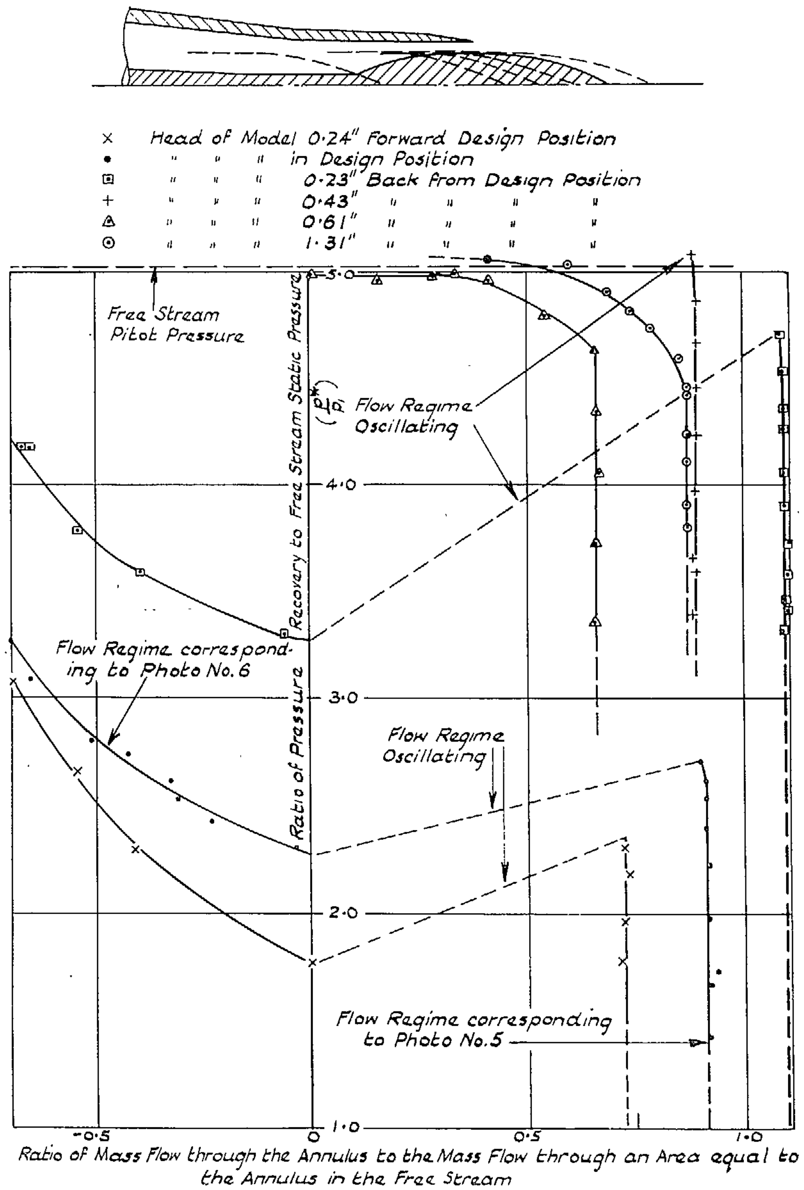


FIG. 9. Variation of mass flow through annular entry (Q1) with pressure recovery at a Mach number of 1·88.

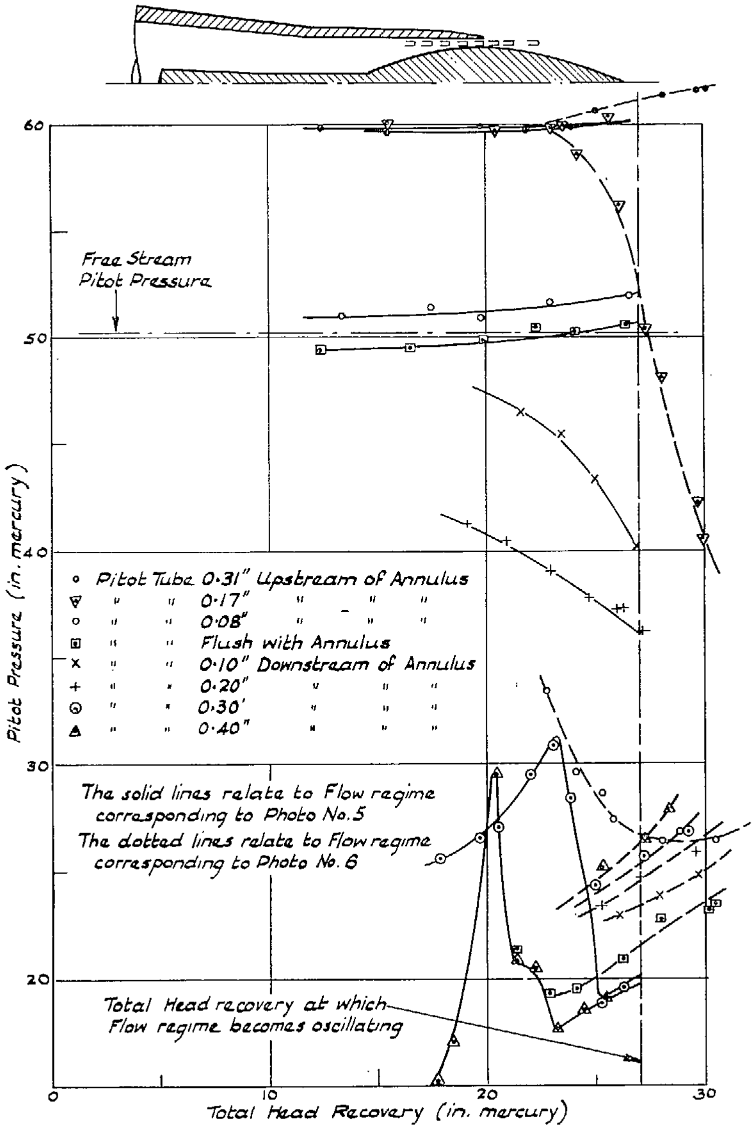
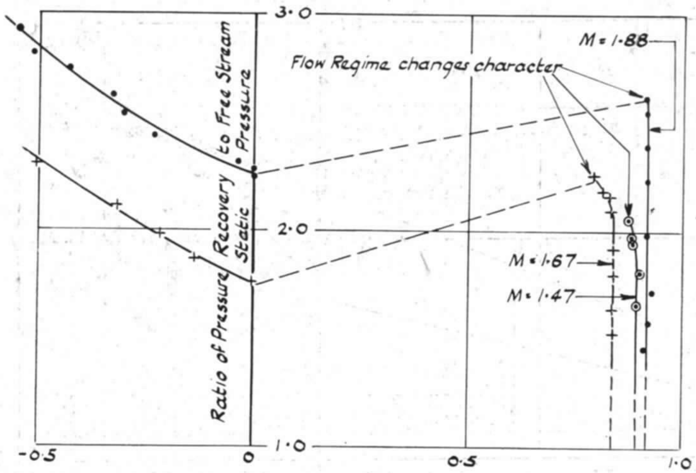


FIG. 10. Variation of pitot pressure near the annulus (Q1) with total-head recovery at diffuser outlet; Mach number 1·88.

(a) Q 1 Entry



(b) E24/43 Entry

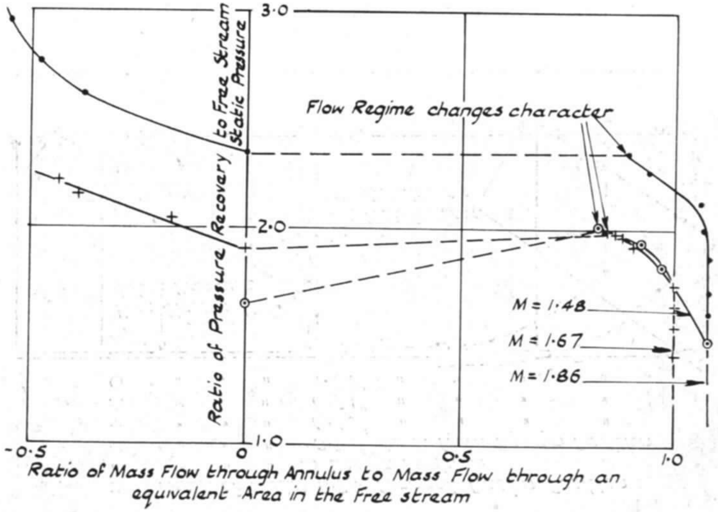
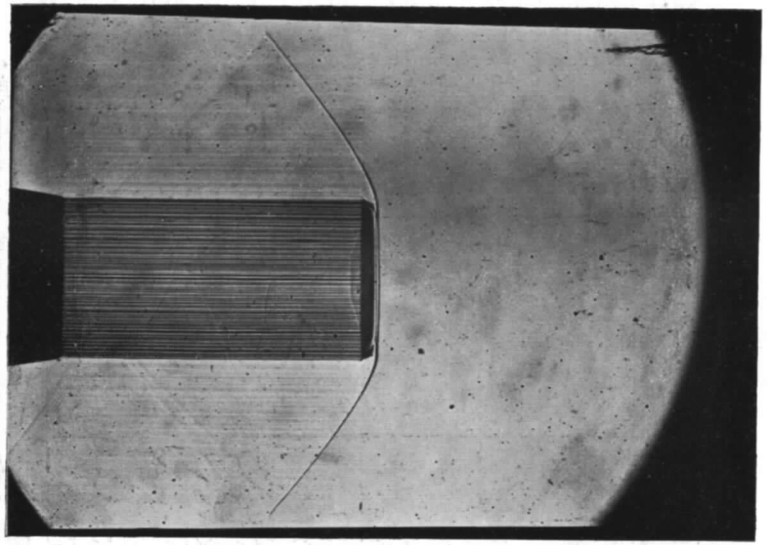
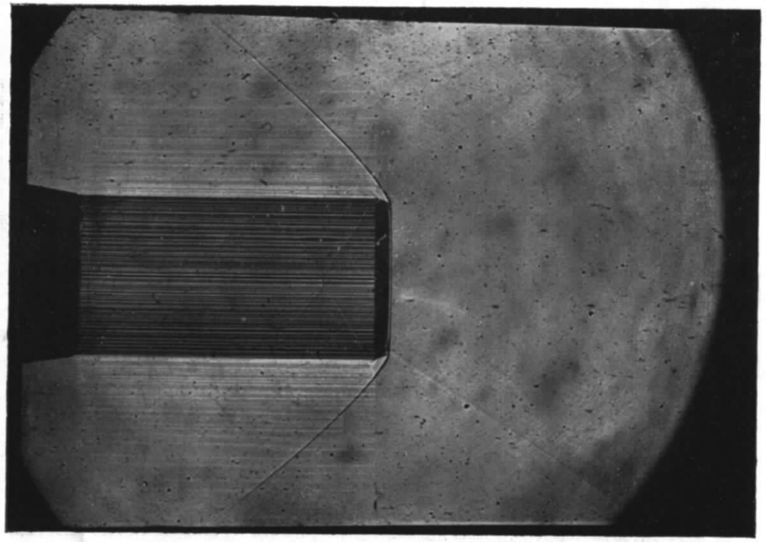


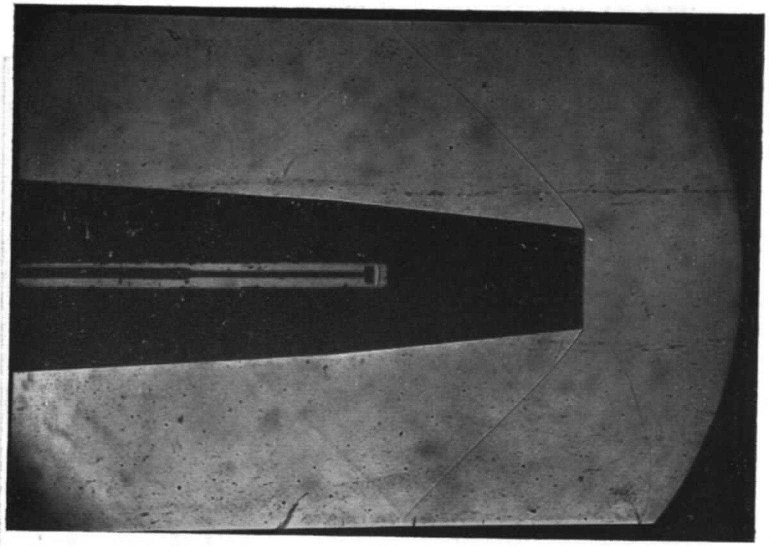
FIG. 11. Variation of mass flow through two annular entries with total-head recovery at several Mach numbers.



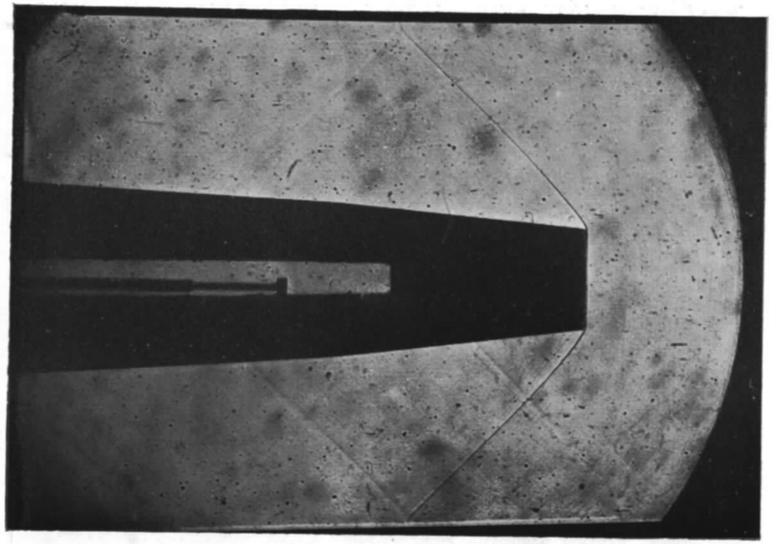
Photograph No. 1. Supersonic flow through a parallel-entry diffuser. $M_1 = 1 \cdot 49$.



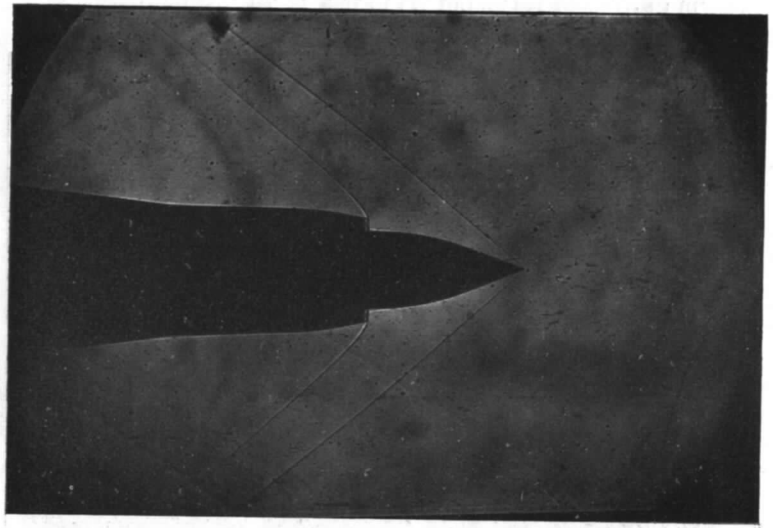
Photograph No. 2. Supersonic flow through a parallel-entry diffuser. $M_1 = 1 \cdot 79$.



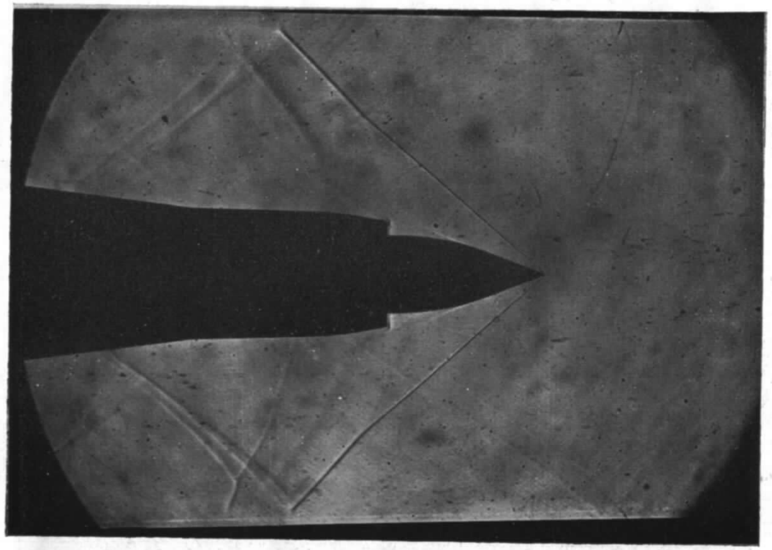
Photograph No. 3. Flow through a uniformly tapered duct (total internal angle 7 deg). $M_1 = 1\cdot45$, $p^*/p_1 \simeq 1\cdot6$.



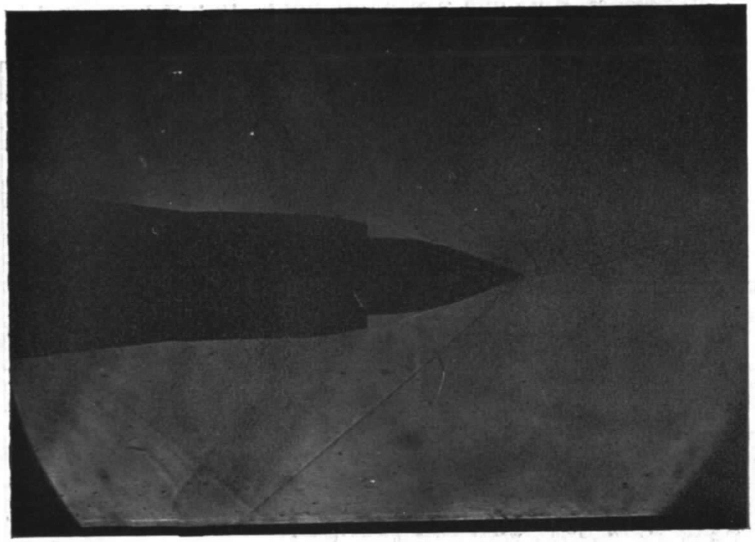
Photograph No. 4. Flow through a uniformly tapered duct (total internal angle 7 deg). $M_1 = 1\cdot45$, $p^*/p_1 \simeq 1\cdot6$.



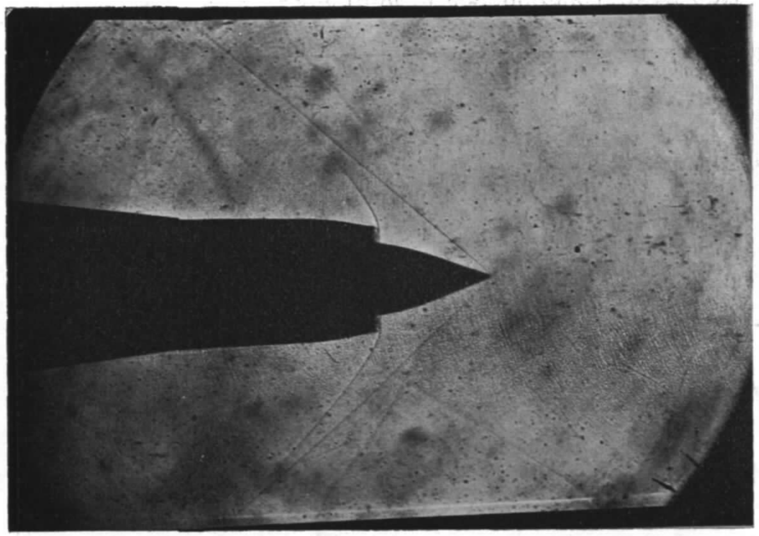
Photograph No. 5. Flow near Q1 annular entry. $M_1 = 1\cdot88$, $p^*/p_1 < 2\cdot7$.



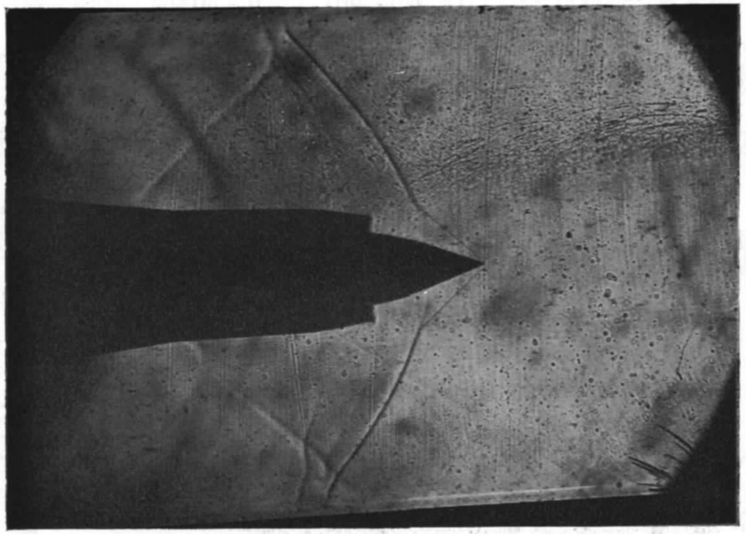
Photograph No. 6. Flow near Q1 annular entry. $M_1 = 1\cdot88$, $p^*/p_1 > 2\cdot7$.



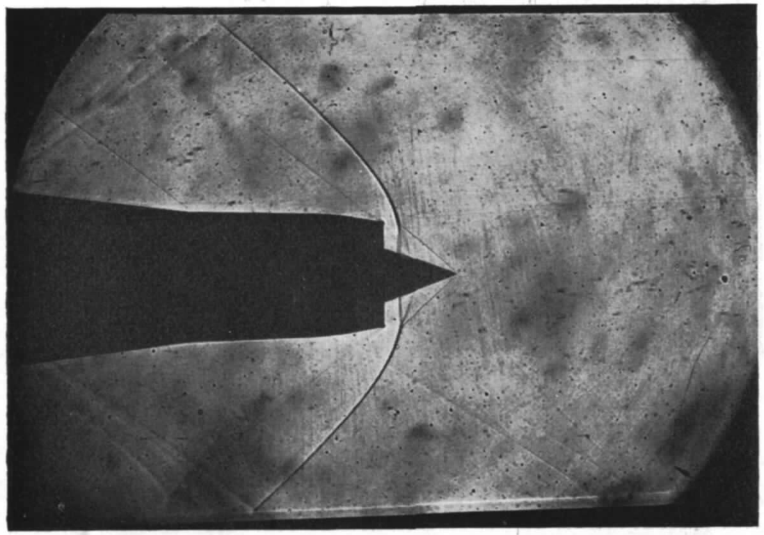
Photograph No. 7. Flow near Q1 annular entry. $M_1 = 1 \cdot 88$.
No flow through the annulus.



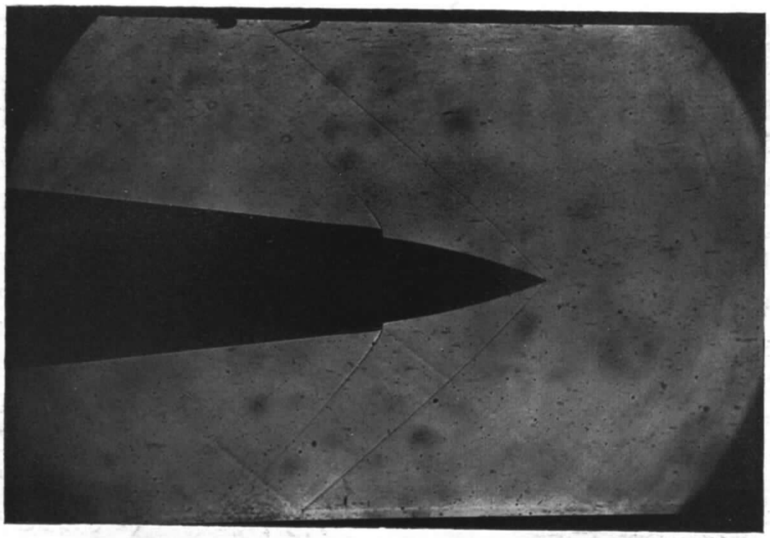
Photograph No. 8. Head of Q1 model set back 0·2 in.
Flow in ' transition ' state. $M_1 = 1 \cdot 88$.



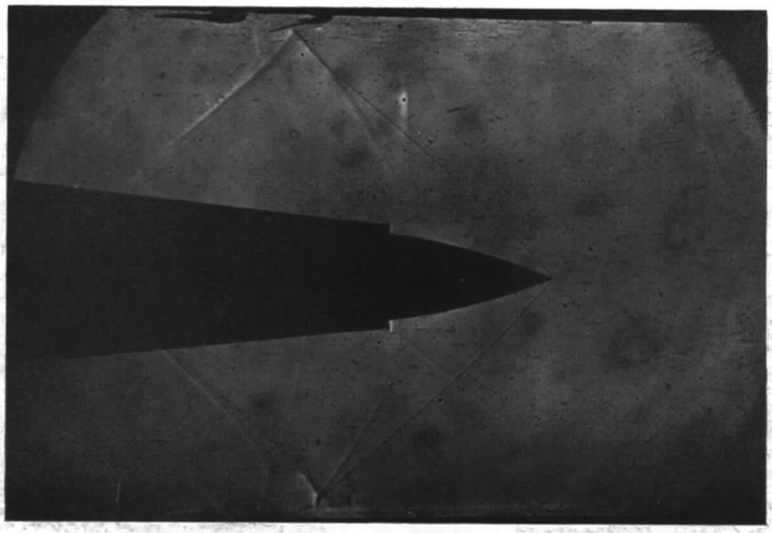
Photograph No. 9. Head of Q1 model set back 0·2 in.
' Reversed ' flow through the annulus. $M_1 = 1 \cdot 88$.



Photograph No. 10. Head of Q1 model set back 0·45 in.
Positive flow through the annulus. $M_1 = 1 \cdot 88$.



Photograph No. 11. Flow near E24/43 annular entry.
$M_1 = 1{\cdot}86$, $p^*/p_1 < 2{\cdot}3$.



Photograph No. 12. Flow near E24/43 annular entry.
$M_1 = 1{\cdot}86$, $p^*/p_1 < 2{\cdot}3$.

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PRINTED IN GREAT BRITAIN

R. & M. No. 282

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