R. & M. No. 2703
(12,527)
A.R.C. Technical Report

MINISTRY OF SUPPLY

AERONAUTICAL RESEARCH COUNCIL
REPORTS AND MEMORANDA

The Improvement in Pressure Recovery in Supersonic Wind Tunnels

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LONDON: HER MAJESTY'S STATIONERY OFFICE
1953
PRICE 4*s* 6*d* NET

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COMMUNICATED BY THE PRINCIPAL DIRECTOR OF SCIENTIFIC RESEARCH (AIR), MINISTRY OF SUPPLY

Reports and Memoranda No. 2703*

May, 1949

Summary.—The inefficient pressure recovery of present day supersonic wind tunnels, which leads to high costs of plant installation and operation, is discussed and methods of improvement suggested. In particular, the diffuser system, where most of the losses occur, is studied in detail; the improvement to be expected in the pressure recovery by the use of convergent-divergent types is explained and methods of overcoming the necessity for high starting powers with this arrangement are presented.

Diffuser experiments based on recent investigations into breakaway phenomena in supersonic flow are described which result in a considerable improvement of pressure recovery. A deceleration from $M = 2 \cdot 48$ at the working section to $M = 1 \cdot 42$ at the diffuser throat was obtained using a variable diffuser throat.

1. *Introduction.*—The inefficient pressure recovery in present day supersonic tunnels is expressed in the large amount of power required to run these tunnels at high Mach numbers, and consequently, in their high building costs.

Large tunnels running at high Mach numbers, as well as small tunnels, having low capital costs, for fundamental research in Universities are in increasing demand. An intensive investigation into the possibilities of an improvement in the pressure recovery is badly needed; this has so far not been carried out.

The aim of this report is to study in detail the various reasons for the inefficiency of the pressure recovery, and to suggest improvements.

Diffuser experiments based on recent investigations into breakaway phenomena in supersonic flow at the Royal Aircraft Establishment are described, which result in a considerable improvement of the pressure recovery. The practical application of such diffusers is discussed.

2. *Pressure Recovery in Supersonic Flow.*—In supersonic flow a pressure recovery by decelerating the flow can, theoretically, be achieved in two ways:—

(a) isentropically, by a suitably designed convergent channel (reversed Busemann nozzle),

(b) non-isentropically, by a shock system.

Since a shock-wave increases the entropy in the flow, a certain amount of the possible pressure recovery is lost. This loss is a maximum for a normal shock, decreases with the deflection angle for an oblique shock, and increases in both cases with the Mach number before the shock.

*R.A.E. Report Aero. 2326, received 18th August, 1949.

An ideal or shockless supersonic diffuser would theoretically decelerate the given supersonic flow to sonic velocity in the throat of a convergent channel (reversed Busemann Nozzle), followed by a divergent subsonic diffuser (Fig. 1a).

In practice, however, the supersonic flow is not initially given but has to be built up from rest in a nozzle. During this unsteady building-up process a shock occurs in the nozzle, the entropy rise of which does not allow the flow to pass through the narrow throat of the diffuser.

In order to establish supersonic flow before the diffuser the throat has to be widened. Consequently, the Mach number obtained at the throat of the diffuser, after the supersonic flow is established, is greater than one. (Details are explained in the next section.)

The change from supersonic to subsonic velocity is now achieved by a normal shock at the throat, followed by subsonic diffusion in the divergent part of the diffuser (Fig. 1b). It follows that a loss in pressure recovery due to a shock wave is unavoidable and that the efficiency of a diffuser has to be considered in connection with the supersonic nozzle. Furthermore, no matter in which way the flow is decelerated in the convergent part of the diffuser, the pressure recovery will be approximately the same for a given Mach number in the working-section, because the size of the diffuser throat and the Mach number there is determined by the building up process only.

In tunnels with an open-jet working-section it may be noted that a second throat is intended, mainly, to adjust the pressure in the working-section. How far this throat may be used for the pressure recovery is not discussed in this report which is concerned only with a closed working-section.

3. *The Building-up Process in a Nozzle Diffuser System.*—A channel with two successive throats (Fig. 2a) may represent the nozzle (throat area A^*), the working-section (area A) and the diffuser (throat area $A^{*'}$) of a supersonic tunnel. One-dimensional flow and $A^* < A^{*'}$ are assumed.

A pressure difference applied across this channel accelerates the flow by means of unsteady pressure waves travelling up and down stream. Assuming that the time in which a pressure difference is applied is large compared with the time taken by the flow to reach a steady state, the problem can be regarded as quasi-steady. Each pressure difference then corresponds to a steady flow state with a velocity distribution such that the sum of the pressure losses due to friction on the wall and entropy losses in shocks equals the applied pressure difference. Fig. 2a shows the pressure distribution for various exit pressures with constant entry pressure.

As soon as the velocity of sound is reached in the nozzle throat (curve b), the mass flow and the flow upstream of the throat is fixed. With further increase in pressure difference the flow expands to supersonic velocity behind the throat and is changed to subsonic flow by a normal shock which travels further downstream with increasing pressure difference. Although the mass flow is fixed, the velocity in the diffuser throat increases because the entropy rise, due to this shock, increases when the shock moves downstream to higher Mach numbers.

If the velocity of sound in the diffuser throat is reached in this way, the flow upstream of the second throat is fixed (Fig. 2a, curve d). A further increase in pressure difference cannot penetrate the sonic second throat and the diffuser acts as a Laval nozzle (curves e and f). Supersonic flow in the working-section cannot be obtained; the tunnel is choked. The area ratio $A^{*'}/A^*$ at which this undesirable state occurs can easily be derived from the continuity equation applied to the nozzle throat (index*) and diffuser throat (index *'), and the fact that the stagnation temperature is constant throughout the system and hence also the sonic velocity and temperature.

If ρ is the density, a the velocity of sound and A the channel area, then

$$\frac{A^{*'}}{A^*} = \frac{\rho^*}{\rho^{*'}} \cdot \frac{a^*}{a^{*'}} \quad \text{(equation of continuity)}$$

and also

$$a^* = a^{*'} \text{ and } \frac{p_0}{p_0'} = \frac{p^*}{p^{*'}} = \frac{\rho^*}{\rho^{*'}} \quad \text{(constancy of sonic temperature).}$$

Hence

$$\frac{A^{*\prime}}{A^*} = \frac{p_0}{p_0{}'} . \quad \ldots \quad (1)$$

It follows that supersonic flow in the working-section is obtained when the velocity of sound at the diffuser throat is avoided, or

$$\frac{A^{*\prime}}{A^*} \geq \left(\frac{p_0}{p_0{}'}\right)_{\max} \quad \ldots \quad (2)$$

is fulfilled during the building-up process. The maximum value of $p_0/p_0{}'$ is obtained for a shock located at the highest Mach number, that is, in the working-section.

If the supersonic flow has built up as far as the working-section and condition (2) is fulfilled (Fig. 2b curve a) any further increase in pressure difference causes the shock to jump from the working-section through the diffuser throat, and to become located in the divergent part of the diffuser at approximately the same Mach number as in the working-section (Fig. 2b curve b). To utilize the gain in pressure recovery by the diffuser, the shock has to be located at the lowest Mach number, which is at the diffuser throat, after the flow has built up. It follows that a certain pressure difference (independent of the diffuser), is required to build up the flow, depending on the Mach number in the working-section only.

Assuming no friction and a normal shock, the ratio of the diffuser throat area to the working-section area, $A^{*\prime}/A$, at which the supersonic flow just builds up, and the Mach number M_{throat} obtained at the diffuser throat after the flow has been built up, may be calculated from equation (2) using the relation between $p_0/p_0{}'$ and the Mach number before a normal shock. In Figs. 3 and 4, $A^{*\prime}/A$ and M_{throat} are plotted against the Mach number in the working section (M_w).

Measurements, by Simons[1], of the ratio $A^{*\prime}/A$ are included in Fig. 3. The agreement with the theoretical curve calculated by assuming a normal shock is accidental because in practice the flow is separated near the wall, and a complex bifurcated shock system, illustrated in Fig. 5, is formed.

4. *The Subsonic Part of the Diffuser.*—The shock at, or downstream of, the second throat determines the flow in the divergent part of the diffuser. The schlieren pictures (Fig. 5) show the formation of shocks in straight-walled divergent channels for various Mach numbers. At low supersonic Mach numbers the shock is nearly normal and a slight detachment of the flow from the walls occurs (Fig. 5, $M = 1 \cdot 2$). At higher Mach numbers the flow is completely detached and the shock bifurcates, (Fig. 5, $M = 1 \cdot 67$ and $2 \cdot 0$). Application of the main body of results obtained from tests on purely subsonic diffusers is therefore not possible.

Tests by A. D. Young[2] and results from Ref. 3 indicate that the pressure recovery is independent of the angle of divergence of the diffuser for angles of up to 8 deg for a Mach number up to $M \approx 1 \cdot 4$ At higher Mach numbers a better pressure recovery is obtained with smaller angles[3]. Systematic tests to determine the optimum angle of divergence over the Mach number range have not yet been done : a total angle of divergence of 5 deg to 7 deg is commonly used.

5. *The Pressure Ratio Requirement in Supersonic Tunnels.*—Instead of defining an efficiency coefficient in terms of the pressure recovery obtained in a nozzle working-section diffuser system, it is more convenient to use the pressure ratio, ψ, required to run the system at a certain Mach number.

Because the velocity of the flow before and after the system considered is usually small, ψ may be approximated by $p_0/p_0{}'$, the ratio of the total pressures before and after the considered system.

In Fig. 6 experimental values for ψ, obtained mainly from tests by Simons[1] are plotted against the Mach number in an empty working-section. Curve a represents the case without a diffuser contraction, that is, with a shock at the end of the working-section. Curve b represents tests by Simons using a second throat adjusted to allow the flow to build up, and placing the final shock as near to the diffuser throat as possible.

Values of ψ without contraction are available for various other tunnels and some of these results are included in Fig. 6. They lie closely to the line through Simons' points despite the range of tunnel sizes.

For comparison, the stagnation pressure ratio across a normal shock is included in Fig. 6: curve a′ corresponds to a normal shock at the working-section; curve b′ corresponds to a normal shock at the diffuser throat, using equation (2) or Fig. 4 to determine the Mach number there.

The curves a′ and b′ represent the pressure recovery loss due to the entropy increase across a shock; the difference between the curves a and a′ and between b and b′ represents losses due to skin friction in the tunnel system and also to dead-water regions in the subsonic part of the diffuser caused by breakaway of flow at the final shock.

In Fig. 7, ψ is plotted against the Mach number before the final shock, with and without diffuser. (The relation in Fig. 4 is used to determine the Mach number at the diffuser throat for curve b.)

The agreement between the two curves shows that it is mainly the Mach number before the final shock which governs the efficiency of a supersonic diffuser.

6. *Improvement of the Pressure Recovery.*—From the previous discussion it follows that there are two ways of obtaining a better pressure recovery with a diffuser :—

(a) By decreasing the Mach number before the final shock at the throat of the diffuser. (Important at high Mach numbers.)

(b) By improving the subsonic pressure recovery. (Important at Mach numbers near 1.)

The Mach number at the throat was determined by the building-up process. A smaller Mach number there may be obtained by :—

(i) Influencing the building-up process so that supersonic flow can be established with a smaller second throat.

(ii) Reducing the throat after the flow has been built up.

No attempt appears to have been made so far to influence the building-up process. One possibility would be to by-pass air by having slots in the convergent part of the diffuser during the building-up process. The decrease in mass flow thus obtained might allow the shock to move through a narrower diffuser throat*. No further suction need be applied after the flow has built up. The problem here is to find out experimentally if the flow can be built up in this way with the final pressure ratio across the system, or if it is essential to have a higher pressure ratio, during the building-up process, which may be provided by applying suction in the divergent part of the diffuser. Reduction of the throat area, after the flow has been built up, can easily be obtained by using flexible walls or hinged rigid walls. An additional pressure ratio to build up the flow is essential in this case. How far the Mach number at the throat can be reduced by narrowing the throat is discussed in the next section.

An improvement in subsonic diffusion may be obtained by influencing the final shock system at the diffuser throat and decreasing the extent of the dead-water region. Too little, however, is known at present about the formation of dead-water regions in connection with a shock wave to make practical recommendations.

For practical application it is important that the diffuser arrangement used is applicable to all Mach numbers and model installations in order to avoid difficult adjustments when the Mach number or model is changed.

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* A vessel evacuated by a small suction pump could be used to provide the suction during the building-up process.

7. *Discussion of the Criterion for the Possible Amount of Contraction of the Diffuser Throat and a Description of the Relevant Tests.*—7.1. *Development of Criterion.*—Once the supersonic flow is built up in a tunnel it is theoretically possible to reduce the diffuser throat until the velocity of sound is reached there. In practice, however, the pressure rise along the wall, or a shock reflected from the wall, will cause the flow to break away before that state is reached. Supersonic flow in the working-section then breaks down because the deflection of the flow from the wall causes an additional shock and hence an additional entropy rise which does not allow the flow to pass the throat. A flow pattern similar to curve d Fig. 2a is then obtained.

The amount the diffuser throat can be reduced and the minimum Mach number which can be obtained there, is limited, therefore, by the onset of breakaway in the convergent part of the diffuser.

With concave-shaped walls, to give shockless compression, the pressure gradient on the wall is larger near the throat, that is, at lower Mach numbers; with convex-shaped walls the pressure rise across the shock reflected on the walls is larger near the beginning of the convergent part, that is, at higher Mach numbers. Therefore, the larger pressure rise at high Mach numbers, obtained with convex-shaped walls, is more favourable for avoiding breakaway.

Further, straight walls inclined at an angle θ can be used (Fig. 8). At each point on the top and bottom walls, or on the side walls where the shocks intersect, the boundary layer is subjected to a pressure rise corresponding to a deflection of flow through two successive oblique shocks each with a deflection angle θ. In Fig. 9, the pressure ratio p_2/p_1 across such a double deflection is plotted against the Mach number before the deflection, for different angles θ.

From recent investigations on breakaway phenomena in supersonic flow (briefly described in Appendix I) an approximate rule was found (Fig. 9, curve a) relating the Mach number of the flow and the minimum pressure ratio p_2/p_1 across a shock or a large local pressure gradient, for which breakaway of a turbulent boundary occurs.

Assuming that this relation can be used for the shock reflection on a wall, the curve (a) determines for each angle θ, a Mach number M_{crit}, such that for $M \leqslant M_{crit}$, breakaway ocurs.

Therefore, if in the convergent part of the diffuser considered the Mach number before each shock reflection or intersection of shocks is greater than M_{crit}, corresponding to, say, θ_1, breakaway can be avoided. This can be achieved by locating the throat so that the Mach number at the last reflection (Fig. 8) or intersection before the throat is equal to, or less than M_{crit}. Supersonic flow is then obtained without breakaway.

7.2. *Description of Tests.*—To prove how far the above reasoning can be put into practice, experiments were carried out in a tunnel with a $5\frac{1}{2}$-in. square working-section, at a Mach number of 2·48, using dry air. Two similar wooden plates were joined at one end by flexible plates to the working-section and chamfered at an angle of 15 deg at the other end to form a throat. The throat width was adjustable by a screw arrangement (Fig. 10); the different plate lengths were 10 in., 14 in., 18 in., 22 in., measured from the working-section to the throat.

The experiments were carried out as follows. First, the flow in the working-section was built up using a large throat area; the throat setting was then decreased, almost to the width where supersonic flow broke down, without actually allowing it to do so. Schlieren pictures were taken; the total head at the throat, the deflection angle of the plates and the throat width were measured.

Any further decrease in the throat area resulted in a sudden and complete breakdown of the supersonic flow; this occurred consistently at the same throat width. During the tests the exit pressure in the tunnel was kept low, so that the flow expanded after the throat.

Schlieren pictures are shown in Figs. 11 to 14. A compression fan originates from the beginning of the convergent part and soon coalesces into an oblique shock. The point of shock reflection is displaced from the wall because of the boundary layer. The observed shock pattern agrees well with theory, provided that the displacement of the reflection point is taken into account. Weak disturbances observed in the picture are caused by the joints of the flexible plate with the wood, and do not affect the flow pattern very much.

Because the limited field of view prevented schlieren pictures of the throat being taken, the Mach number there was obtained by extending the shock pattern, making an approximate allowance for the displacement of the reflection points (Figs. 11 to 14).

Test Results.—The results given are for the limiting conditions just prior to breakdown of supersonic flow in the working-section, using the different diffuser plate lengths.

Length of convergent part of diffuser (in.) (length L, Fig. 8)	10	14	18	22
$\dfrac{L}{\text{Width of working-section}}$	1·82	2·54	3·27	4·0
Diffuser throat width, $A^{*\prime}$ (in.) (width of nozzle throat, $A^* = 2\cdot09$ in.)	3·27	3·02	2·76	2·56
$\dfrac{p_t}{p_0} = \dfrac{\text{Pressure actually recorded by total head tube at diffuser throat}}{\text{Stagnation pressure before the nozzle}}$	0·725*	0·865	0·893	0·915
$\dfrac{p_0'}{p_0} = \dfrac{\text{Stagnation pressure at diffuser throat}}{\text{Stagnation pressure before the nozzle}}$	0·988	0·975	0·968	0·960
Deflection angle of diffuser wall θ deg.	6·4	5·07	4·26	3·65
M_{crit}	1·97	1·79	1·66	1·54
Mach number at diffuser throat, M_{throat}	1·74	1·62	1·52	1·42

* p_t/p_0 was measured at M_{crit} in this case (*see* Fig. 14).

The pressure losses in the convergent part due to friction and shocks were derived as follows :—

If p_0 is stagnation pressure in working-section,

p_0' stagnation pressure at diffuser throat,

M_t Mach number at diffuser throat,

p_t pressure actually recorded by a total-head tube placed at the throat,

then the loss of total pressure in the convergent part of the diffuser can be expressed as a ratio of the stagnation pressure in the working-section and can be written

$$\frac{p_0 - p_0'}{p_0} = 1 - \frac{p_0'}{p_0} .$$

Now,

$$\frac{p_0'}{p_0} = \left(\frac{p_0'}{p_t}\right)\left(\frac{p_t}{p_0}\right) .$$

The stagnation pressure ratio p_0'/p_t across the normal shock in front of the total-head tube, is a function of M_t only and is a known quantity, and p_t/p_0 can be measured; hence p_0'/p_0 can be calculated.

The measured width of the throat is larger than the width calculated from M_{throat} because of the displacement thickness of the boundary layer. (The boundary layer on the side walls has to be considered as well.)

In Fig. 9 the relationship between p_2/p_1 (the static pressure ratio across a reflected shock) is plotted against Mach number for different angles θ. The measured values of M_{crit} are plotted for comparison with the estimated M_{crit} (curve a). The increasing discrepancy between the estimated and measured values for small angles, θ, could be explained by the difference in the state of the boundary layer at M_{crit} for the two cases. Curve a was obtained by using the boundary layer at the working-section; the points obtained from the diffuser tests depended on the state of the boundary layer at the diffuser throat, which had been affected by several shock reflections.

In general, it can be said that the test results agree well with the theoretical explanation of the problem, and that curve a may be used as a rough guide to estimate the Mach number obtainable at the throat of the diffuser.

The following estimation of the pressure ratio required to run a tunnel at $M = 2 \cdot 48$, using the test results, shows the considerable gain in pressure recovery which can be obtained by using a diffuser with a contraction. In the estimate, the final shock is located in the divergent part some distance after the throat, at a higher Mach number than M_{throat}, in order to prevent any influence of the final shock acting upstream through the boundary layer.

Working-section Mach number	2·48
Mach number at diffuser throat using a convergent part 4 times the width of the working-section	1·42
Final shock located at a Mach number of	1·60*
Pressure ratio required for $M = 1 \cdot 6$ according to Fig. 6..	1·50
Pressure ratio to account for the friction loss in the convergent part $p_0'/p_0 = 0 \cdot 9$ assumed	1·11
Required pressure ratio with convergent diffuser..	1·68
Required pressure ratio without convergent diffuser (Fig. 6)	2·80

A further improvement of the pressure recovery might be obtained by having convex diffuser walls, instead of the straight ones used in the experiment, because the boundary layer there is accelerated between the shock reflections.

With a model in the working-section the pressure recovery is decreased by only a small amount; disturbances, consisting of shocks and expansions from the model, have almost cancelled each other out before they reach the critical region near the throat, and therefore, should not affect M_{crit} appreciably.

In practice the setting of the throat width to suit any combination of Mach number and model in the working-section is easily achieved with the suggested variable throat diffuser.

8. *The Provision of an Additional Pressure Ratio in Supersonic Tunnels.*—In section 6, two ways are suggested for obtaining a smaller Mach number at the diffuser throat. The use of a diffuser throat has been shown by the experiments described to be feasible; the use of suction as a means of influencing the building-up process requires confirmation before practical application is possible.

Provision of the greater initial pressure ratio required to build up the flow, before the throat can be narrowed, is now discussed.

Two tunnel systems have to be considered separately.

(a) *Intermittent tunnels* where the discharge from a high-pressure vessel, or the flow into an evacuated vessel, or the discharge of a high-pressure vessel feeding an injection tunnel, is used to obtain supersonic flow during a short period of time. In the interval between each run the air in the vessel is brought back to its original state.

* This allows an arbitrary increase, over the minimum value of 1·42, to ensure that the subsonic diffuser flow does not interfere with the flow in the convergent part of the diffuser.

(b) *Continuously running tunnels* using a compressor in an open or closed tunnel circuit.

One way of improving an intermittent tunnel would be to increase the duration of each run. For tunnels using a vessel evacuated to the initial pressure p_i, the expression for the maximum running time, $t_{\max}$, is

$$t_{\max} = \frac{1}{\gamma}\left(\frac{\gamma+1}{2}\right)^{(\gamma+1)\,2(\gamma-1)} \frac{1}{a_0}\,\frac{1}{A^*/A} \cdot \frac{V}{A} \cdot \left(\frac{p_{t\,\max}}{p_0} - \frac{p_i}{p_0}\right)$$

where

$p_{t\,\max}$	vessel pressure just before the breakdown of the flow,
V	volume of vessel,
A^* and A	the cross-sectional areas of the nozzle throat and working section respectively,
p_0 and a_0	the pressure and velocity of sound, respectively, for the stagnation conditions.

Without reducing the diffuser throat the pressure distribution is similar to Fig. 2b. The flow over-expands in the divergent part of the diffuser because the pressure ratio p_0/p_i is larger than that required to build up the flow (Fig. 2b curve d). As the pressure in the vessel increases the final shock moves upstream towards the diffuser throat, and supersonic flow breaks down if the throat is reached (Fig. 2b curve c), that is if $p_0/p_{t\,\max}$ equals the required pressure ratio Ψ (Fig. 6). A similar equation for $t_{\max}$ can be derived for the case of a high pressure vessel and an injection tunnel.

It follows that a variable diffuser throat can be applied to intermittent tunnels. According to the above formula for $t_{\max}$ a decrease in ψ, by reducing the throat, results in an increase in running time at a given Mach number or an increase in the maximum Mach number obtainable in the working-section. To obtain hypersonic Mach numbers for a reasonable running time and tunnel size such a diffuser is essential.

In a continuously running tunnel the maximum possible Mach number obtainable in the working-section is determined by the pressure-ratio–volume intake characteristics of the compressor, the size of the working-section and the pressure ratio required to build up the flow. Using a variable diffuser, therefore, the maximum Mach number is not increased unless an additional pressure ratio is provided during the building-up process.

Even without an additional pressure ratio, the use of a variable throat diffuser decreases considerably the power required to drive the compressor and to cool the air during the continuous running. Furthermore, using the same amount of power, the air density in the circuit can be increased which is especially desirable at higher Mach numbers.

To provide an additional pressure ratio, suction, or injection of compressed air behind the diffuser, may be applied. The practical usefulness of this arrangement has yet to be proved experimentally.

This problem of providing a large pressure ratio for starting can be circumvented if the main tunnel nozzle upstream of the working-section can be made adjustable in some way. For example, if a flexible walled nozzle is developed which can be operated while the tunnel is running or if a sliding nozzle of the type developed by the N.A.C.A. were used. It should be emphasized that for this purpose the nozzle need not vary in such a way as to provide a uniform airstream at each stage during the starting or accelerating process, and that a simple hinged nozzle or a crudely shaped sliding nozzle might be acceptable. Possible arrangements are sketched in Fig. 15.

In all these cases flow is built up at a lower Mach number, corresponding to the available pressure ratio, by either increasing the nozzle throat width or decreasing the working-section width. The walls of the nozzle working-section and the diffuser are then adjusted to the final shape.

During this process the width of the diffuser throat is adjusted so that the Mach number there does not exceed the Mach number at which the flow was originally built up. This arrangement is most promising for future tunnels.

9. *Conclusions*.—It is shown that in a supersonic diffuser, that is, a convergent-divergent channel of fixed geometry, the deceleration in the convergent part of the diffuser is limited because a certain minimum width of the throat is required to enable the building-up of supersonic flow.

Therefore, the Mach number before the shock at the diffuser throat, where the flow changes from supersonic to subsonic, is greater than unity, leading to a high pressure-ratio requirement because of the increasing losses in pressure recovery with increasing Mach number before a shock.

Several recommendations for decreasing this pressure-ratio requirement are made, of which the use of a variable diffuser throat after the flow has been built up is the simplest for practical application.

A criterion is developed showing the limitation of the possible deceleration in the convergent part of the diffuser by the occurrence of breakaway. Experiments with a variable diffuser throat are described and a deceleration from $M = 2 \cdot 48$ at the working-section to $M = 1 \cdot 42$ at the diffuser throat was obtained.

By using a variable diffuser throat in an intermittent tunnel the running time at high Mach numbers is increased, or for the same running time a considerably higher Mach number is obtained at the working-section as compared with a fixed diffuser.

In a continuously running tunnel, a variable diffuser throat decreases the amount of power required during the running, or for the same amount of power the density in the circuit can be increased. If an additional pressure ratio is provided to build up the flow, the Mach number obtainable at the working-section is increased by a variable diffuser throat. Using flexible walls an increase in the Mach number of the tunnel is possible without an additional pressure ratio.

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APPENDIX I

Extract from an Investigation into Breakaway Phenomena in Supersonic Flow

An experimental investigation into breakaway phenomena occurring in two-dimensional supersonic flow was undertaken at the R.A.E. The experiments were stopped at the exploratory state; only Mach numbers larger than $1 \cdot 85$ are covered and details of the condition of the boundary layer are not known.

The main investigation dealt with the formation of dead-water regions at the rear end of a model, and in front of an obstacle mounted on a flat plate.

Behind a step (Fig. 16a and Fig. 18) the flow expands from the initial Mach number M_i, to a Mach number M_d, enclosing a dead-water region in which the static pressure p_d, corresponds to M_d. The flow is then deflected along the plate by a compression fan which merges into an oblique shock. Fig. 16b shows the pressure distribution along the plate. The peak pressure on the plate equals the initial static pressure before the expansion, indicating isentropic compression near the wall. In front of a step (Fig. 17 and Fig. 19), the flow breaks away at a certain angle causing an oblique shock. In both cases the initial state of the boundary layer is turbulent; the case of the laminar boundary layer need not be considered in this report.

The Mach number to which the flow expands round the corner at the back step, or the angle the flow deflects in front of the step, cannot be determined from the equations for flow neglecting viscosity. In either case the flow was found to adjust the pressure in the dead-water region to a definite value, which for larger step heights depends mainly on the Mach number. At A in Fig. 16, and B in Fig. 17, the boundary layer has to bear a certain local pressure rise. The fact that the flow adjusted this pressure rise automatically led to the rough rule that if the boundary layer in supersonic flow is affected by a smaller local pressure rise than this, breakaway does not occur.

The pressure ratio p_2/p_1, which the boundary layer is able to bear, may be obtained as a function of the Mach number by varying the angle of plate deflection ε deg behind the step (Fig. 18). In Fig. 20, p_2/p_1 is plotted against the Mach number. Two nozzles of $M = 1 \cdot 85$ and $M = 2 \cdot 48$ were used; the nozzle end was the corner round which the flow expanded. The values of p_2/p_1 for the front step are included and show good agreement, although the turbulent boundary layer there was obtained in a different way, *i.e.*, by spoilers on the plate (Fig. 19). It was found that the relation $p_2/p_1 = 1 + CM^2$ (dotted curve in Fig. 20) agreed well with the measured values, where $C = 0 \cdot 25$. The constant C in this relation was found to depend generally on the height of the step as well as on the condition of the boundary layer.

With increasing height at the back step, the pressure in the deadwater region decreases to an asymptotic value. The steps used to obtain Fig. 20 were large enough to be in the asymptotic region. The dependence of p_2/p_1 on the boundary-layer thickness was not so critical in the turbulent as in the laminar case.

Finally, there are indications that the effect of a pressure rise on a turbulent boundary layer, provided, for example, by an oblique shock reflection, causes the flow to break away if the value of p_2/p_1 (Fig. 20) is exceeded. Tests to confirm this statement in detail could not be carried out.

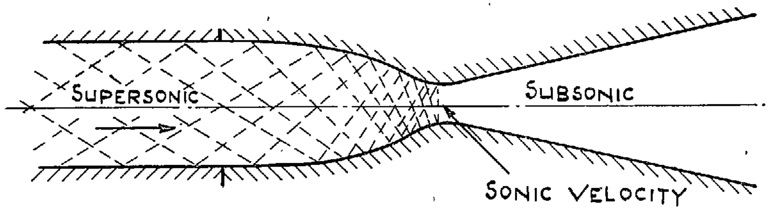


FIG. 1a. Ideal theoretical supersonic diffuser.

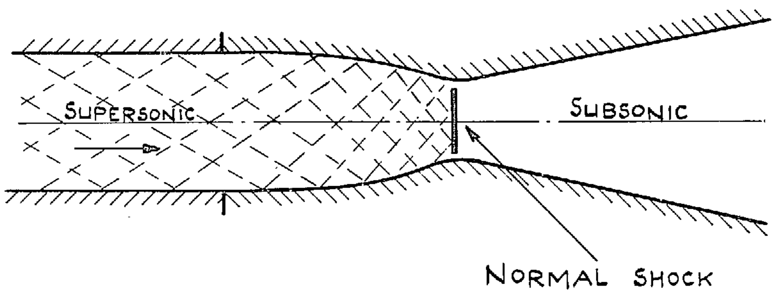


FIG. 1b. Practical supersonic diffuser.

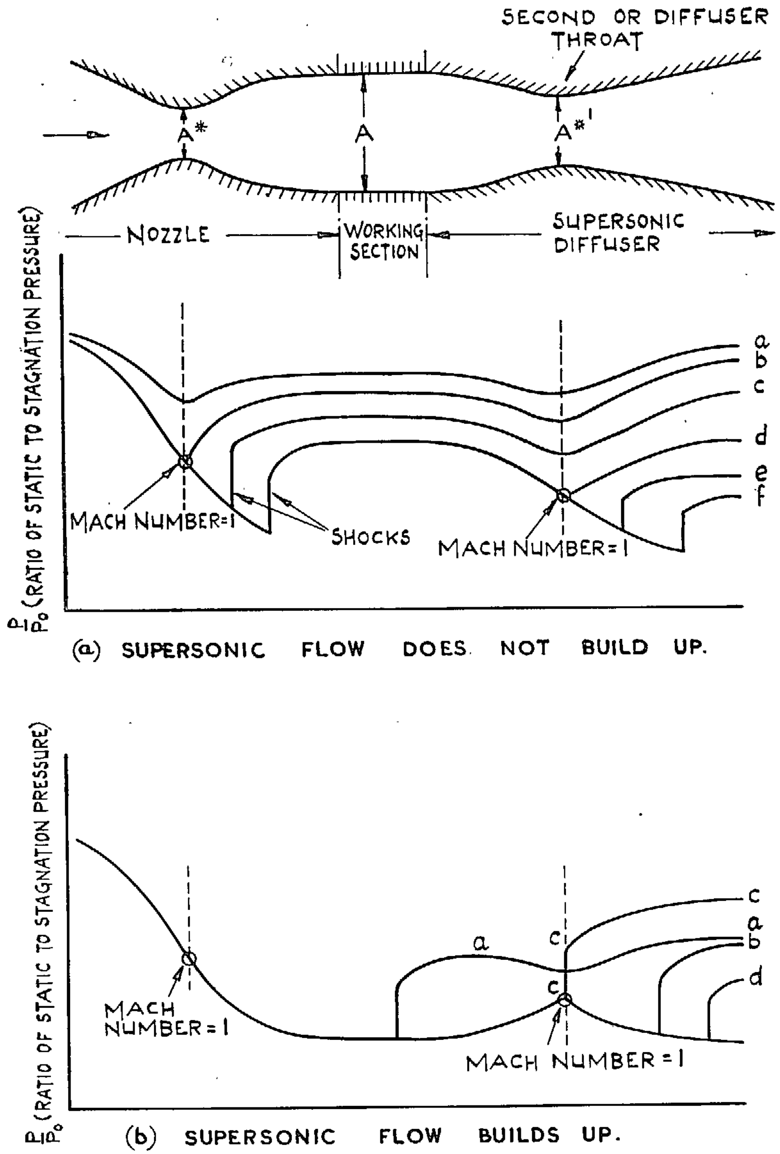


FIG. 2. Pressure distribution along a tunnel during the build-up of supersonic flow.

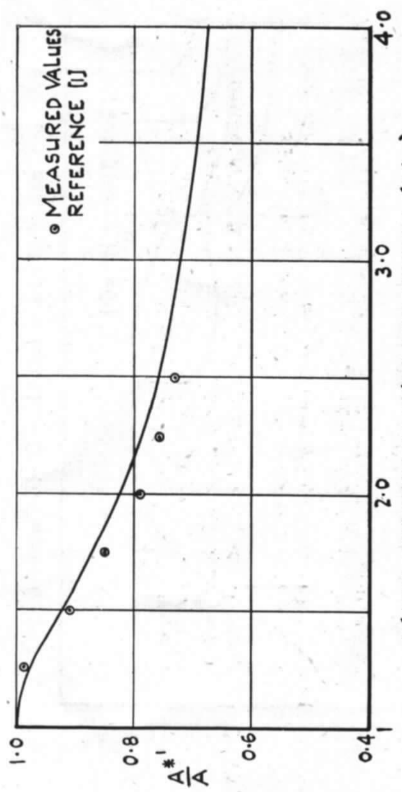


FIG. 3. Relationship between the minimum value of $A^{*\prime}/A$ for which supersonic flow can build up in the working-section and M_w.

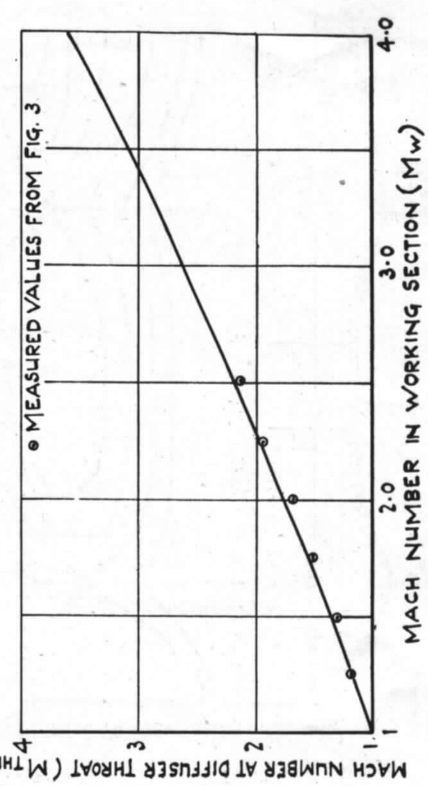
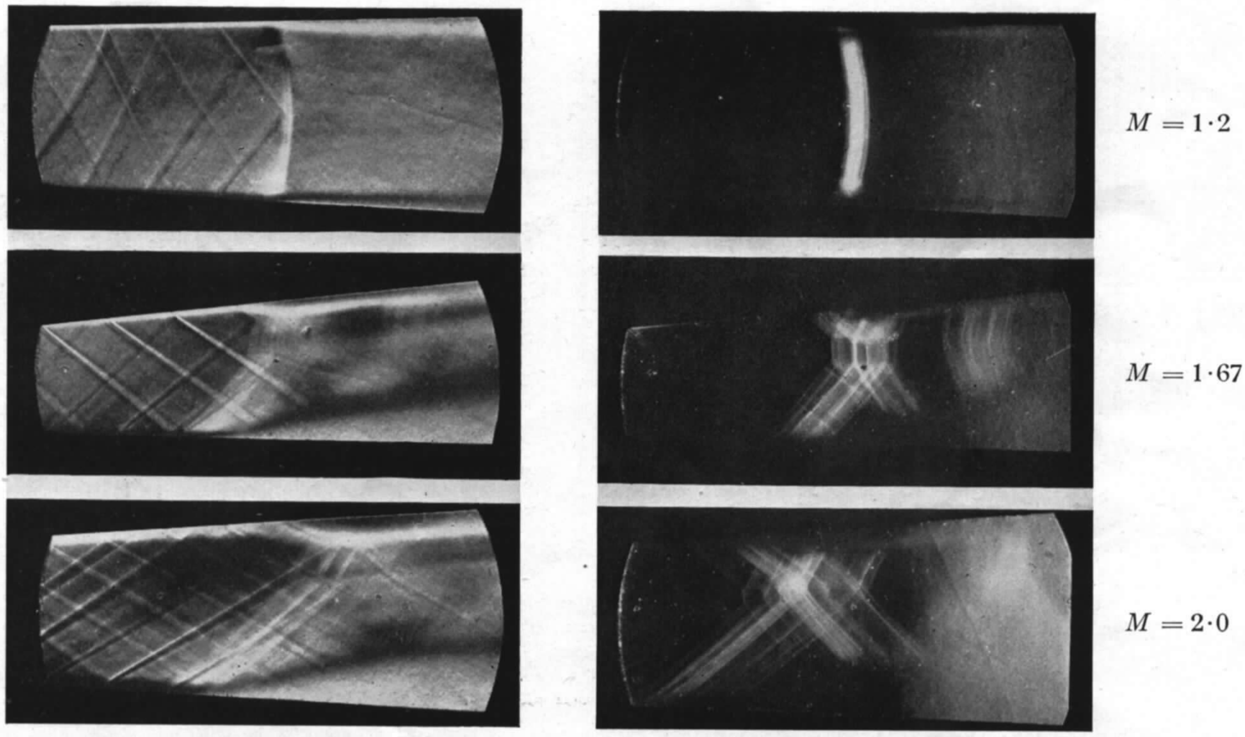


FIG. 4. Relationship between M_{throat} and M_w for the minimum value of $A^{*\prime}/A$ for which supersonic flow can build up in the working-section.



(a) Knife-edge horizontal. (b) Knife-edge vertical.

FIG. 5. Schlieren photographs of shocks in a divergent channel.

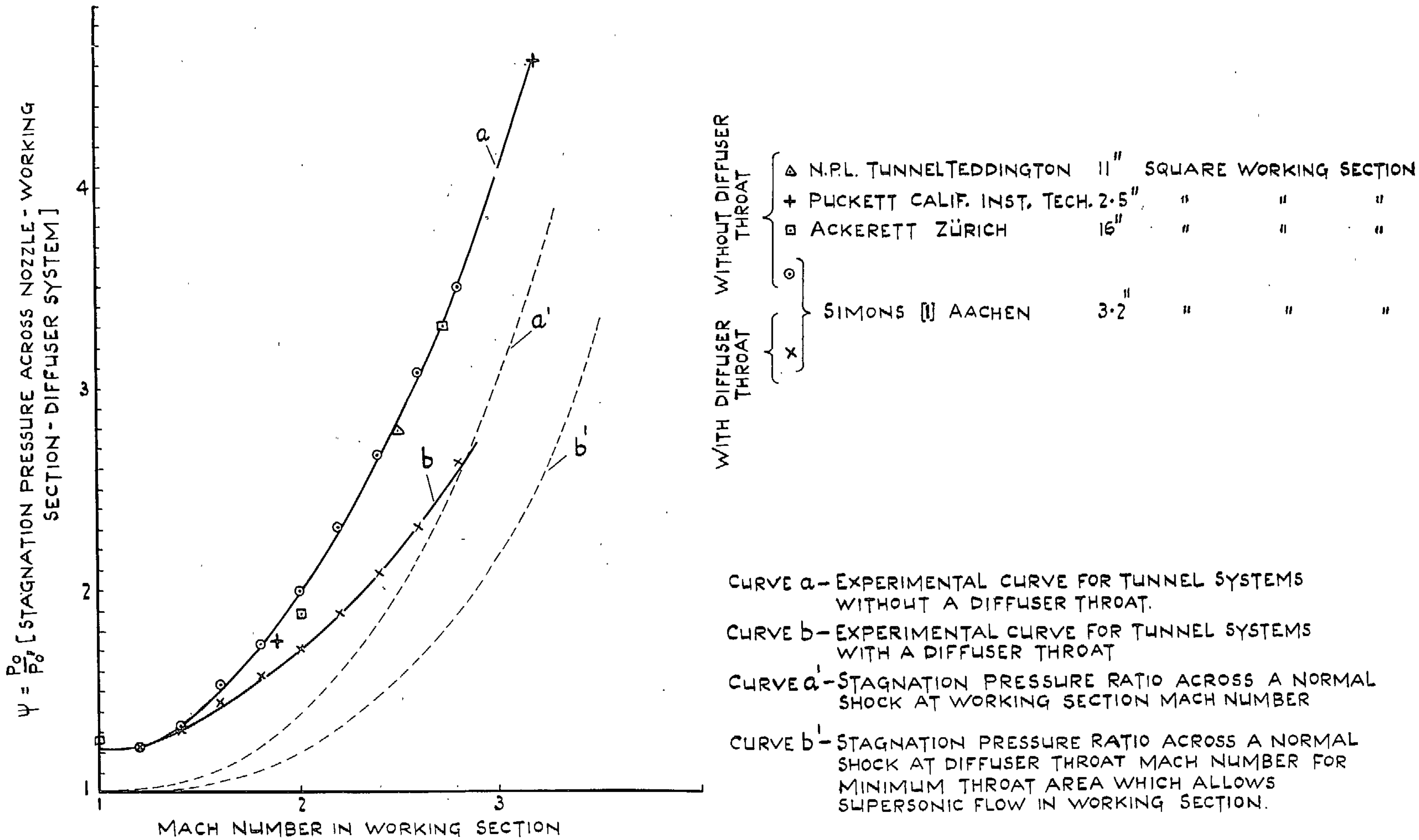


Fig. 6. Relationship between the Mach number in the working-section and the necessary pressure ratio Ψ, across the nozzle–working-section–diffuser system for closed jet tunnels.

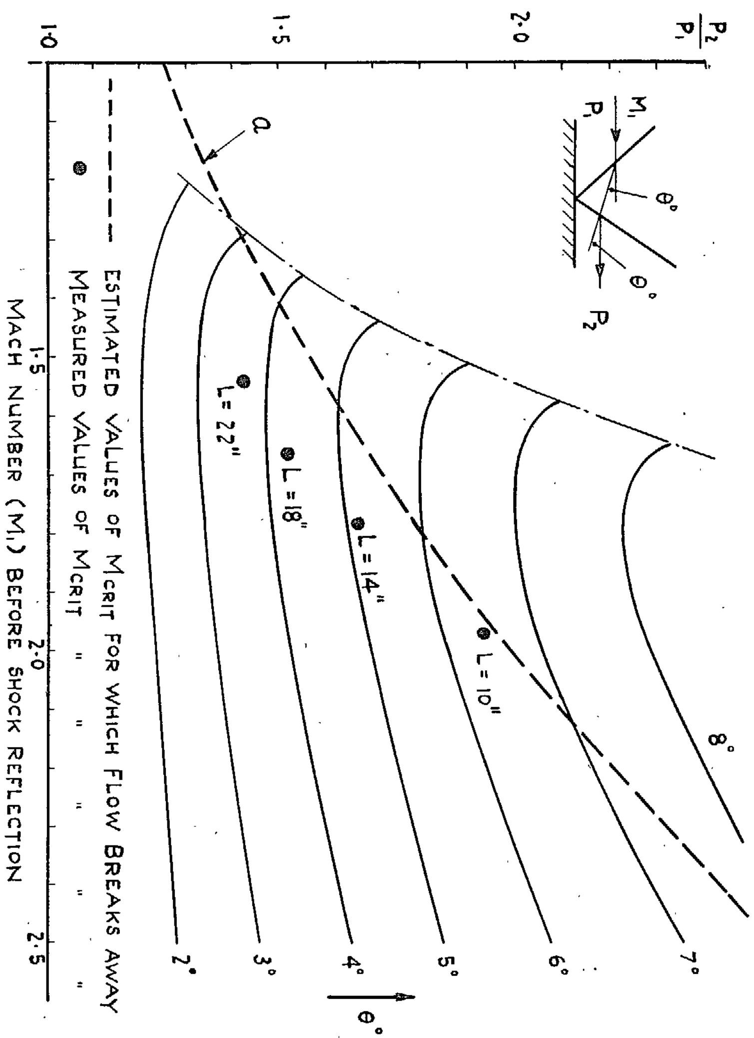


FIG. 9. Pressure ratio across a shock reflection on a wall.

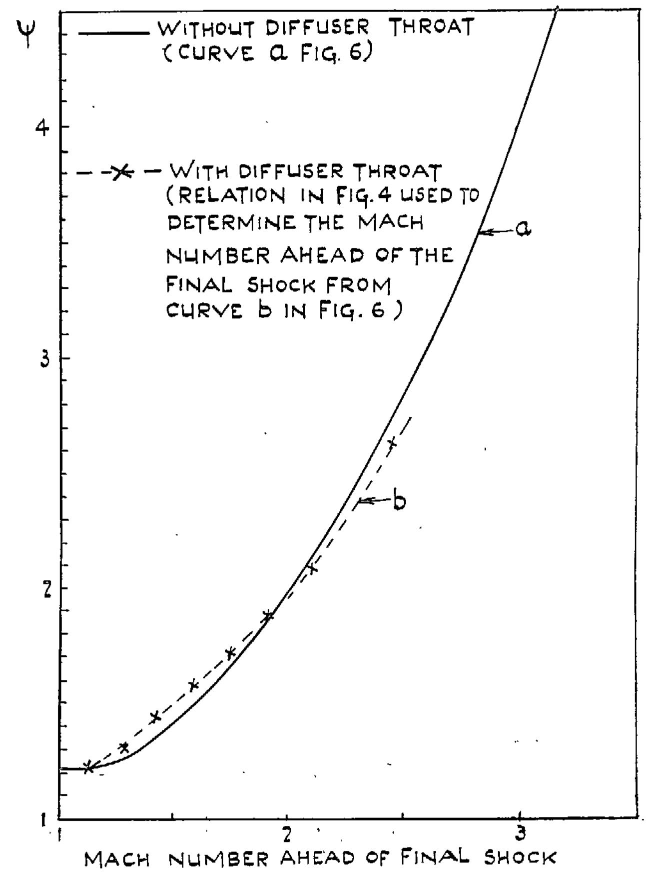


FIG. 7. Required pressure ratio Ψ against Mach number ahead of final shock without and with diffuser throat.

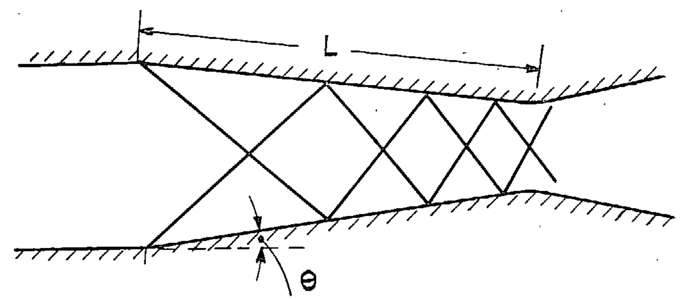
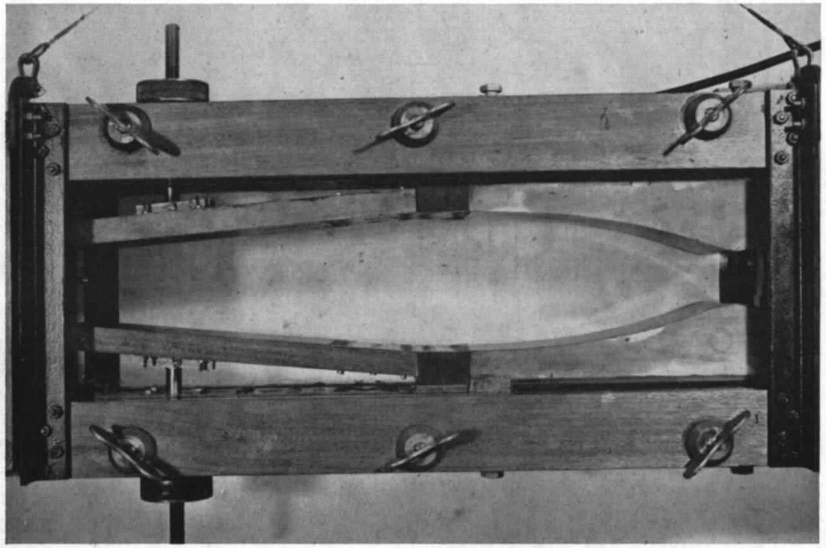
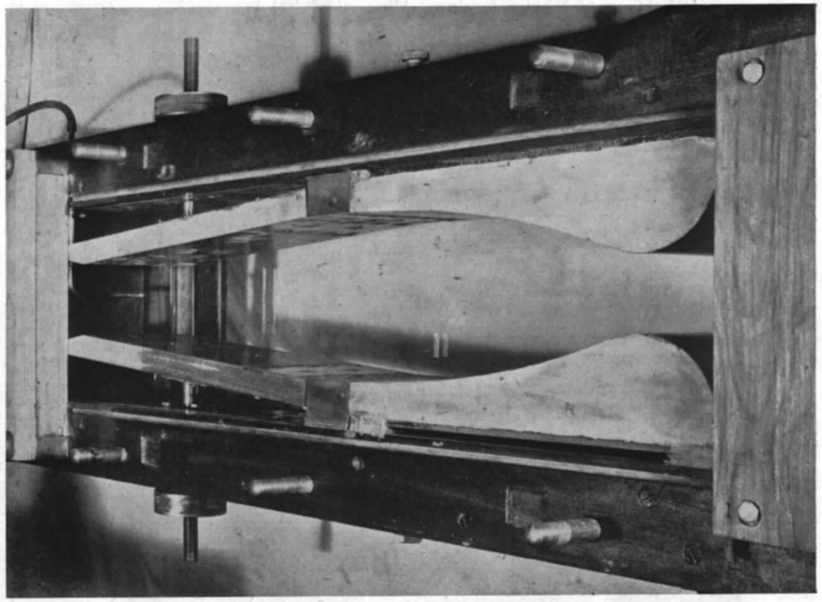


FIG. 8. Straight-walled convergent diffuser.





(a) FIG. 10. Test arrangement. (b)

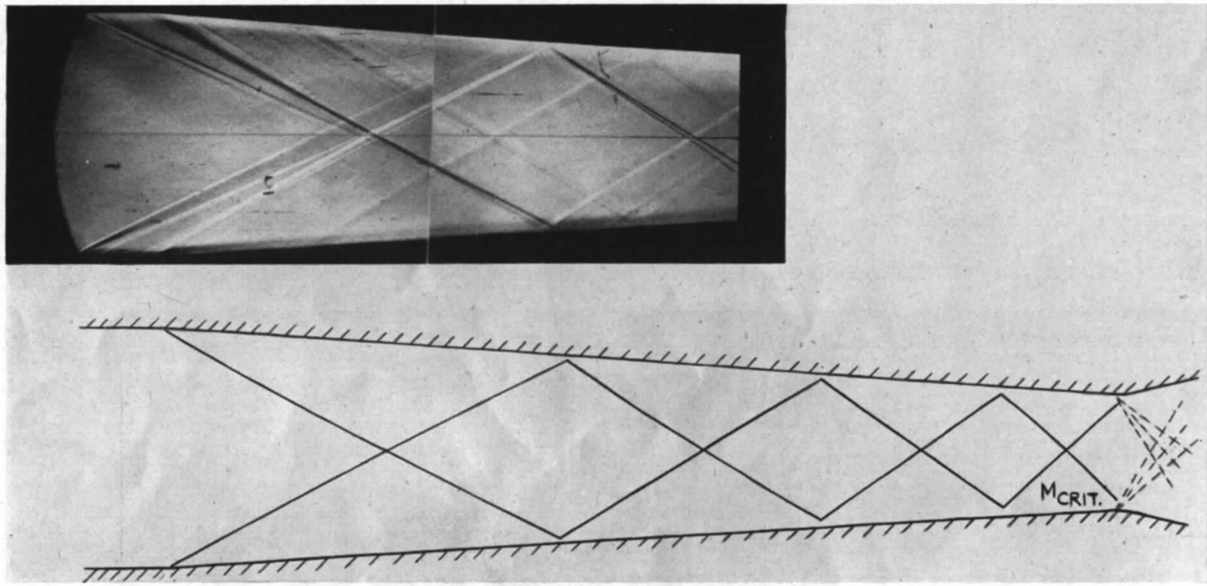


FIG. 11. Diffuser. Length $L = 22$ in.

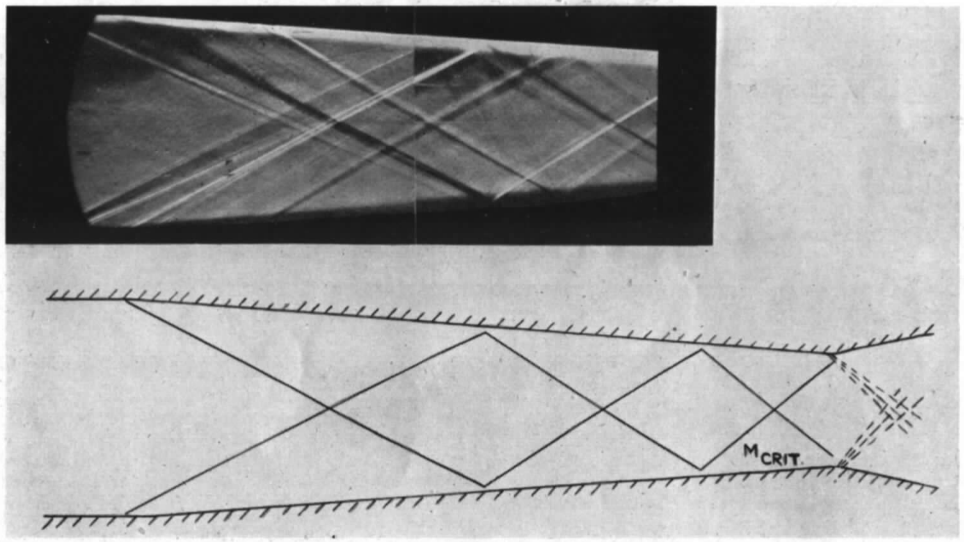


FIG. 12. Diffuser. Length $L = 18$ in.

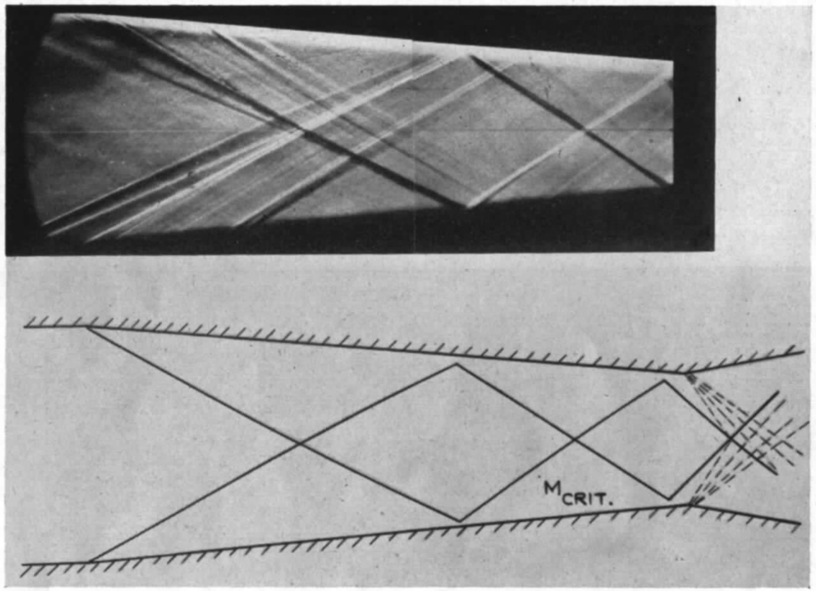


FIG. 13. Diffuser. Length $L = 14$ in.

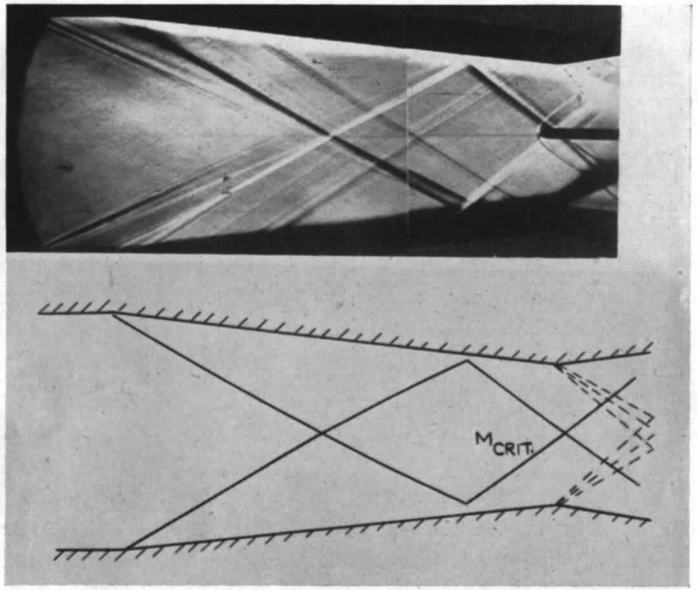


FIG. 14. Diffuser. Length $L = 10$ in.

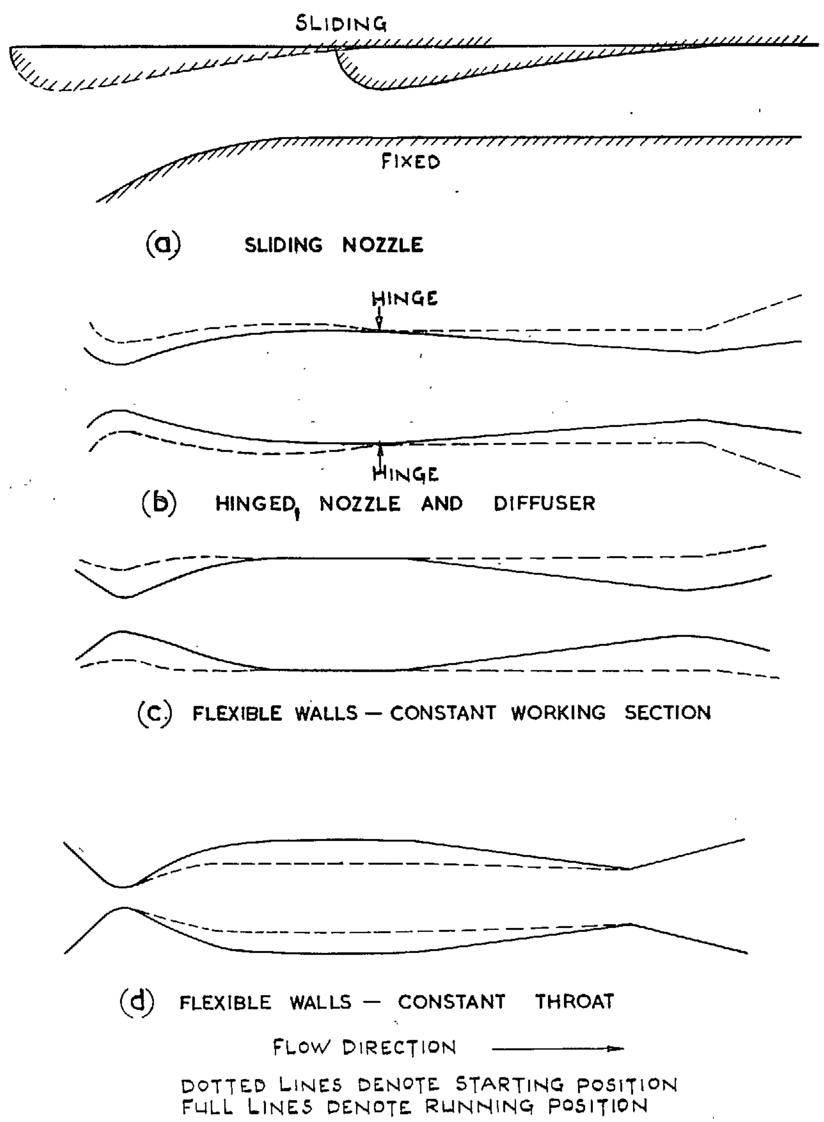


FIG. 15. Possible methods of obtaining supersonic flow in a tunnel without the use of extra power for starting.

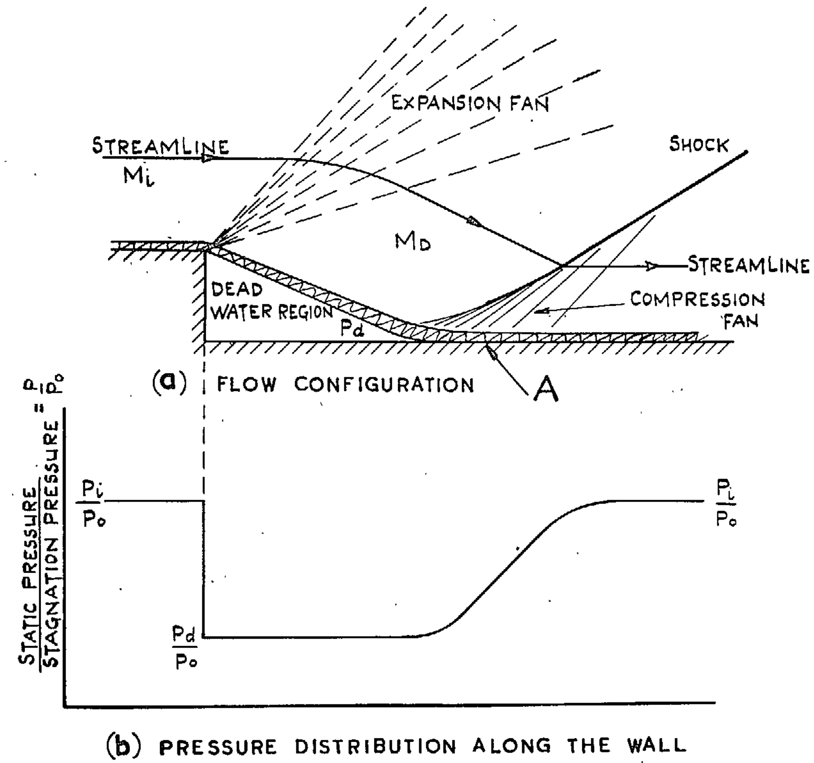


FIG. 16. Flow behind a step.

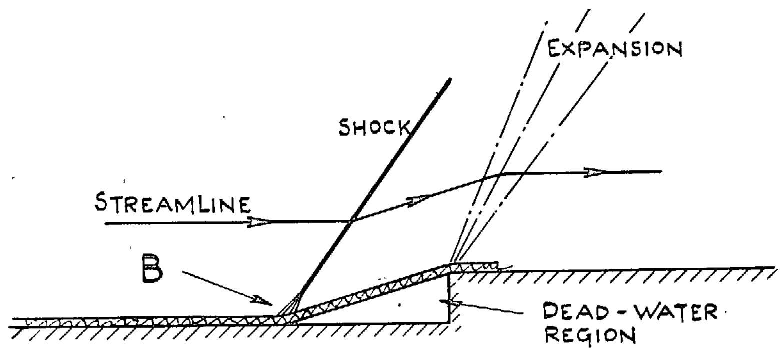


FIG. 17. Flow in front of a step.

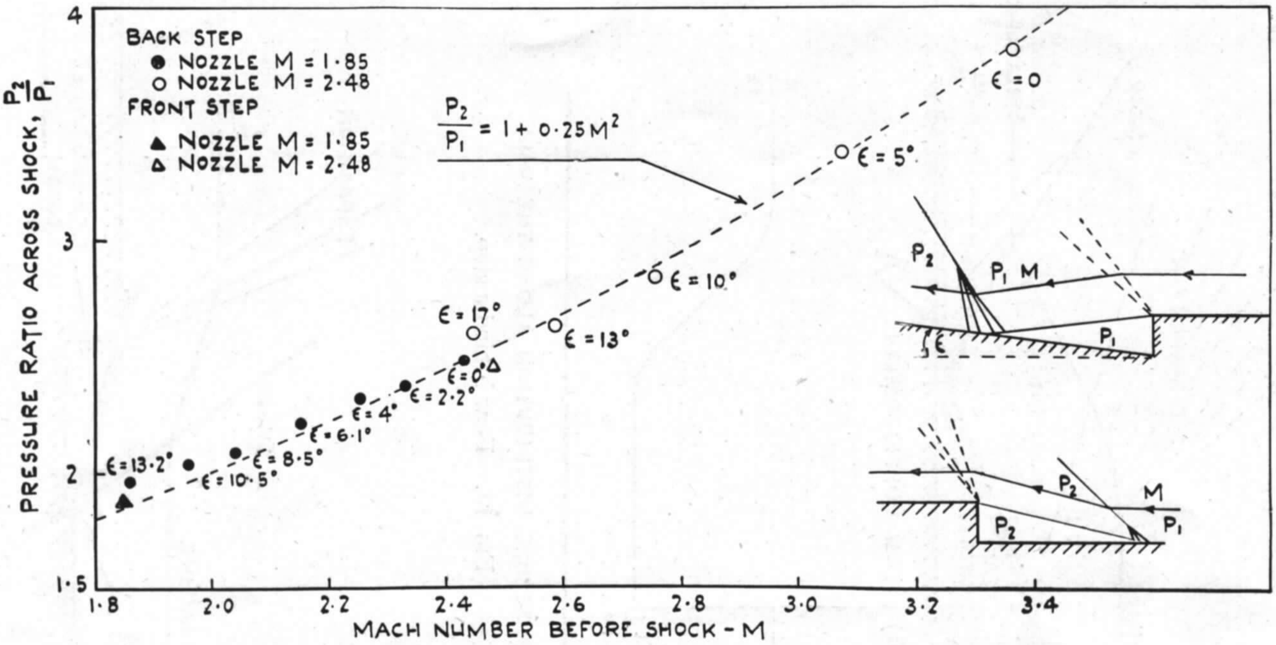


FIG. 20. Breakaway with a shock. Relationship between the pressure ratio across the shock and the Mach number before the shock.

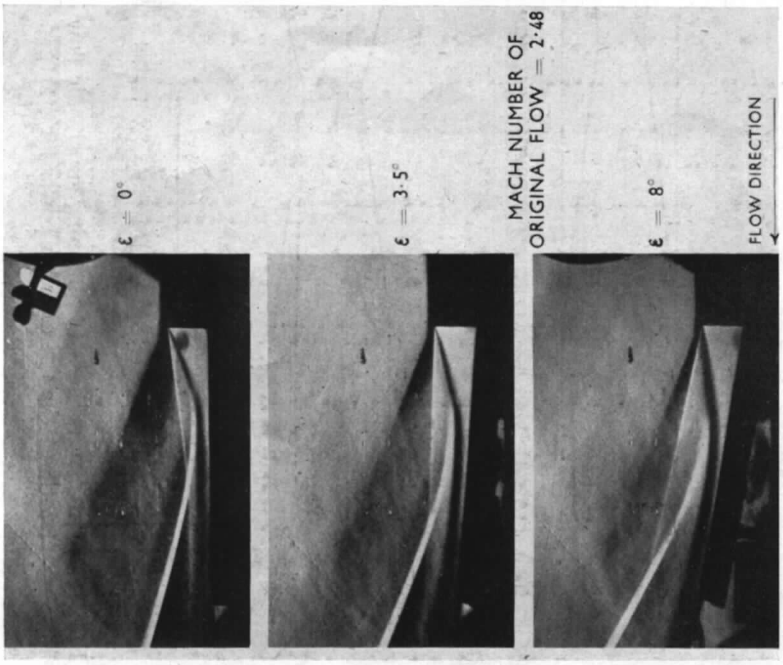


FIG. 18. Schlieren photographs of back step for various angles ε.
(ε = angle of back plate to original flow direction)

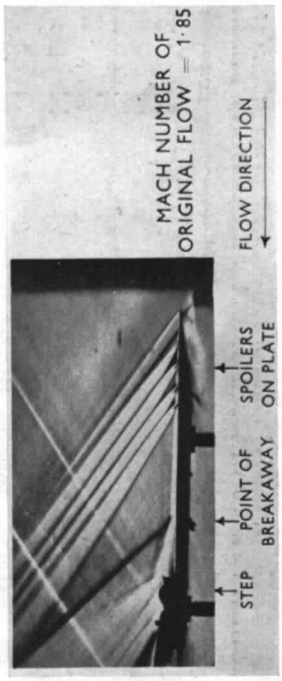


FIG. 19. Schlieren photograph of front step.

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