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by

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AN EXPERIMENTAL STUDY OF THE GLANCING INTERACTION BETWEEN A SHOCK WAVE AND A TURBULENT BOUNDARY LAYER

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SUMMARY

An experimental study has been made at Mach numbers from 1.6 to 2.0 of the interaction between the turbulent boundary layer on a side wall of a wind tunnel and the shock wave produced by a plate mounted on the wall. Under these conditions the shock wave/boundary layer interaction was three dimensional at least over the region investigated (up to 10 boundary layer thicknesses from the plate). It was found that the boundary layer was separated by a shock wave of strength $p_2/p_1 \doteqdot 1.5$, i.e. a flow deflection between 7.5° and 8°. Interactions of this type occur on the sides of fuselages at the wing/fuselage junction and may therefore be important with regard to the design of waisted shapes.

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1 INTRODUCTION

It has been shown[1] that the boundary layer on the side of a fuselage may separate as the flow approaches the wing. At subsonic speeds this is caused by the adverse pressure gradient in the flow up to the stagnation point at the root. At supersonic speeds it is dependent on the pressure rise through the bow shock wave of the wing; if the shock is attached to the wing, and the flow deflection angle (arising from wing thickness and incidence) is small, separation does not occur; while if the flow deflection angle is large separation does occur. This type of shock-induced separation is investigated further in the present Note. It is a problem which is of importance in the design of waisting for wing-fuselage junctions, and on the side walls of rectangular intake diffusers.

Although considerable work has been done on the interaction of shock waves with boundary layers, most of it has been done in cases where the change in flow direction through the shock wave in in a plane normal to the surface. Here we are concerned with the case where the basic flow deflection through the shock wave lies in a plane parallel to the surface on which the boundary layer interaction is being studied. This case is referred to here as a "glancing" interaction between a shock wave and a boundary layer.

The present investigation was undertaken to provide information on the shock strength sufficient to cause separation, and to study the type of flow which occurs in these conditions.

2 EXPERIMENTAL DETAILS

The tests were made in a 4 inch × 4 inch tunnel[2] at nominal Mach numbers of 1.6, 1.8 and 2.0 at Reynolds numbers of 0.37×10^6, 0.35×10^6, and 0.33×10^6 per inch, respectively.

The shock-producing plate, which was mounted normal to the sidewall turntable, is shown in Fig.1. The leading edge angle was chosen to ensure an attached shock wave on the wedge surface when the flat surface was in line with the stream at $M = 1.6$. Nineteen static pressure holes were provided in the surfaces of the wall and plate (Fig.1) to show fine details of the flow near the junction, such as the vortices of Ref.1. The pressures were measured on multitube mercury manometers.

The nature of flow on the sidewall in the presence of the shock wave was observed by means of a surface oil flow technique[3], using titanium dioxide as pigment. The resulting patterns were photographed through the window during tunnel running. The oil used was fairly fluid with little tendency to evaporate (Shell Limea 931, see Ref.3) and it was found to be possible to obtain more than one flow pattern during a run (at least as far as the behaviour of the boundary layer at the shock wave was concerned). This aspect of the technique is discussed in the Appendix. All except one of the flow patterns used in the report to illustrate the types of flow were obtained as the first pattern in a run, i.e. they could not have been affected by patterns produced earlier in the run.

Transition of the boundary layer on the walls of the tunnel was fixed, throughout the tests, by means of threads upstream of the throat. Pitot pressure traverses through the boundary layer showed it to have approximately 1/5-power-law velocity profile and thickness 0.14 inch.

3 PRESENTATION OF THE RESULTS

Pressure distributions on the wall and the plate and a selection of photographs of the oil flow patterns obtained are presented in Figs.2 and 3 for $M = 1.6$, Figs.4 and 5 for $M = 1.8$, and Figs.6 and 7 for $M = 2.0$.

The pressure distributions were obtained at even values of the flow deflection angle together with some additional readings at odd values near the flow deflection angle at which the boundary layer began to separate from the bottom wall. To facilitate visual interpretation of the variation with deflection angle the pressure distributions at odd values of the deflection angles are drawn as broken lines. The theoretical pressure rises through shock waves for the deflection angles are also shown in Figs.2, 4 and 6, together with the theoretical positions of the shock waves and of the Mach waves from the root leading edge, and the actual positions of separation obtained from the oil flow patterns (see section 4).

The photographs were taken with the optical axis of the camera slightly below the plane of the shock-producing plate so they show a perspective view of the undersurface of the plate with a reflection, in it, of the oil flow on the lower half of the wall. The tip section may be seen, slightly out-of-focus, just above the junction of the plate and the wall in the photographs. The oil flow lines have a wavy form because the photographs were taken at a relatively early stage in their development. This is discussed further in the Appendix. The theoretical positions of the shock waves are indicated on each photograph by means of lines extended outside the photographs (see Fig.3). Deflection angles are quoted both for the shock on the flat lower surface of the plate (δ_L) and for the shock on the wedge upper surface of the plate (δ_u). These deflections are relative to the nominal stream direction since no measurements have been made of the flow inclination in the tunnel. A study of the results from the upper and lower surfaces in the present tests suggest that this flow inclination is very small.

4 DISCUSSION OF THE RESULTS

Before discussing the present results it is useful to recall the mechanism of the two-dimensional interaction which occurs when the change in flow direction through the shock wave is in a plane normal to the surface. This is described by Holder, et alia, in Ref.4 and illustrated in Fig.8.

If the incident shock is of only moderate strength, with a deflection angle of less than about 6°, flow separation does not occur. There is then a very steep rise of pressure (Fig.8(a)) at the point where the shock strikes the boundary layer and conditions are similar to those for regular reflection of a shock at a plane wall.

With stronger shocks, separation occurs (somewhat upstream of the point where the shock strikes the boundary layer) and the pressure distribution at the wall is similar to that shown in Fig.8(b). The pressure rises steeply up to the separation point after which the pressure gradient falls (since the rapidly thickening dead-air region cannot withstand a large adverse pressure gradient). Downstream of the point where the shock wave strikes the boundary layer the pressure gradient increases, but it decreases again after reattachment, which occurs close to the peak pressure position.

A two dimensional shock wave/boundary layer interaction of one or other of these forms occurred on the bottom wall of the tunnel throughout the tests (Figs.3,5,7). At small deflection angles the effect on the pressure measurements is negligible but at large deflection angles the separations become sufficiently large to affect the measured pressure distribution. This may be observed beginning at $\delta_L = 7°$ in Fig.2 (for M = 1.6), at $\delta_L = 11°$ in Fig.4 (for M = 1.8), and at $\delta_L = 13°$ in Fig.6 (for M = 2.0). At these angles and above it is difficult to distinguish side wall effects from bottom wall effects in the pressure distributions.

Examination of the oil flow patterns (Figs.3,5 and 7) obtained in the present investigation shows that the surface flow is affected some distance ahead of the shock wave, this distance increasing with distance from the shock-producing plate. Thus the shock wave/boundary layer interaction is essentially three-dimensional in this case, at least over the limited region investigated (about 10 boundary layer thicknesses from the plate). It is likely that an asymptotic condition would be reached at much greater distances from the plate. The wall pressure holes, being in a plane perpendicular to the shock-producing plate and some distance downstream of its leading edge, cross the shock wave (at an angle) between a quarter and a third of the distance from the plate to the bottom wall of the tunnel. The measured distributions therefore give a qualitative indication of the streamwise pressure distribution.

The present results follow a pattern similar to that of the two-dimensional interaction. At small deflection angles (see, for instance, $\delta_L = 6^\circ$ at M = 2.0 in Fig.7(a)) the shock wave is not strong enough to separate the boundary layer and the oil flow lines are merely turned through a greater angle than the streamlines outside the boundary layer by the transverse pressure gradient. However, in contrast with the two-dimensional case, there is significant upstream influence; the pressure begins to rise and the oil flow is deflected well upstream of the shock. As already noted the upstream effect increases with distance from the plate and it commences at a line which is well in advance of the shock front. This could be explained by a form of spanwise influence, the effects of the presence of the shock wave being propagated along the rearward facing local Mach cones which, in a substantial part of the total boundary layer thickness, have angles greater than the shock wave angle. Thus, the effects would be spread more near the bottom wall, where Mach waves are received from all parts of the shock wave/boundary layer intersection, than near the plate, where none are received. A similar process would occur within the subsonic part of the boundary layer. At still greater distances from the plate (in the absence of the bottom wall) it is expected that the interaction would assume a cylindrical form as variation of the spanwise influence decreases.

At much larger deflection angles (see, for instance, $\delta_L = 12^\circ$ at M = 2.0 in Fig.7(d)) the occurrence of separation associated with the presence of the shock wave is obvious. The "incident" oil flow lines are deflected towards a line originating at the root leading edge and lying well upstream of the shock wave. Further downstream the oil flows downwards from behind the shock wave towards this same line. It would appear that the boundary layer separates at this line and rolls up to form a vortex which trails downstream. The presence of a vortex is consistent with slight suction peaks in the pressure distribution on both wall and plate (at about 0.4 and 0.2 inch from the junction, respectively) and scouring action of a vortex would produce downward flow away from the plate.

Between these two cases it is difficult to assign a particular deflection angle as the value at which separation begins. This has, in fact, been chosen fairly arbitrarily as the angle at which the oil flow line from the root leading edge is swept at the same angle as the shock wave. This definition ensures that in the separated case there is "upstream" flow normal to the shock wave while in the attached case there is not.

The oil flow patterns have been analysed on this basis and the results are shown in Figs.9 and 10 as boundaries, of deflection angle and corresponding theoretical overall pressure ratio against Mach number, for which the boundary layer begins to separate. The pressure at separation varies with deflection angle at each Mach number (see Figs.2,4,6). The ratio of the lowest value of the pressure at separation to the free stream static pressure is also shown in Fig.10 for $M = 1.8$ and 2.0 (at $M = 1.6$ the corresponding pressure distribution appears to have been affected by the bottom wall interaction). It will be seen that separation first occurs for deflections of 7.5° to 8° (parallel to the surface) and overall pressure ratios of about 1.5, at local pressure ratios of 1.32.

Comparison with the value of pressure ratio at separation in two dimensional flow[4] (also shown in Fig.10) shows that separation occurs at much lower pressure ratios in the present case.

5 CONCLUSIONS

It has been found that a turbulent boundary layer on the side wall of a wind tunnel will be separated by a glancing shock wave of strength $p_2/p_1 \doteqdot 1.5$ at Mach numbers from 1.6 to 2.0. This pressure ratio corresponds to a flow deflection of between 7.5° and 8°.

Within the region investigated, i.e. up to 10 boundary layer thicknesses from the shock producing plate, the interaction is of marked three-dimensional character. At greater distances from the plate the interaction would be expected to approach asymptotically a cylindrical form.

LIST OF SYMBOLS

H	total pressure
M	Mach number
p	static pressure
p_1	static pressure upstream of shock wave
p_2	static pressure downstream of shock wave
δ_L	deflection angle on lower surface of wedge
δ_u	deflection angle on upper surface of wedge

LIST OF REFERENCES

No.	Author(s)	Title, etc
1	Stanbrook, A.	Experimental observation of vortices in wing-body junctions. A.R.C. R.& M.3114. March, 1957.
2	Stanbrook, A., Secomb, D.	The flow around a rod passing longitudinally through an asymmetric supersonic nozzle. Unpublished M.O.A. Report.
3	Stanbrook, A.	The surface oil flow technique as used in high speed wind tunnels in the United Kingdom. A.R.C.22,385. August,1960. Will be published as an Agardograph.
4	Holder, D.W., Pearcey, H.H., Gadd, G.E., Seddon, J.	The interaction between shock waves and boundary layers, with a note on the effects of the interaction on the performance of supersonic intakes. A.R.C. C.P.180. February, 1954.

APPENDIX 1

OBSERVATIONS OF OIL FLOW PATTERNS IN MORE THAN ONE CONDITION DURING A SINGLE TUNNEL RUN

By the use of a fairly fluid, non-evaporating oil it was found to be possible to obtain oil flow patterns at more than one flow deflection angle during a single run. The particular point of interest in the present investigation was, of course, the occurrence of separation due to the presence of the shock wave. Consequently, the individual patterns were allowed to develop only to the point at which the type of flow at the shock wave could be determined. This was found to require about six minutes.

A typical set of photographs is shown on the left of Fig.11. The first oil flow pattern of the run was that shown in Fig.7(d). The second, third and fourth patterns obtained were those shown in Fig.11. The third pattern was obtained at $\delta_L = 10^\circ$ and on the right hand side of Fig.11 is shown a pattern obtained at the beginning of a separate run. Comparison of these two photographs shows qualitatively the same type of flow with the separation line in the same position. There are some differences in the patterns on the downstream side of the separation line. In the left hand picture this region had been previously scoured by the vortex flows associated with the separations at $\delta_L = 12^\circ$ and 14°. Most of the oil would have been removed from this region and it would consequently take much longer to alter the oil flow pattern there. This effect may also be seen in the lowest photograph, taken at a deflection angle at which separation does not occur. In this photograph the streamwise oil flow lines cut across the feint pattern produced earlier in the run.

It is obvious, therefore, that this use of the oil flow technique is acceptable for the limited object of determining whether or not separation (or any other clearly definable aspect) occurs in a particular region. The technique was found to be most satisfactory when alternate patterns showed separated and attached flows, i.e. when the differences between consecutive patterns was large.

Four consecutive patterns shown in Fig.12 indicate a tendency for the oil filaments to have a wavy form in the early stages of formation. (While rotation of the turntable altered the deflection of the plate and, hence, the oil flow pattern on the turntable, the basic flow upstream of the turntable remained unaltered.)

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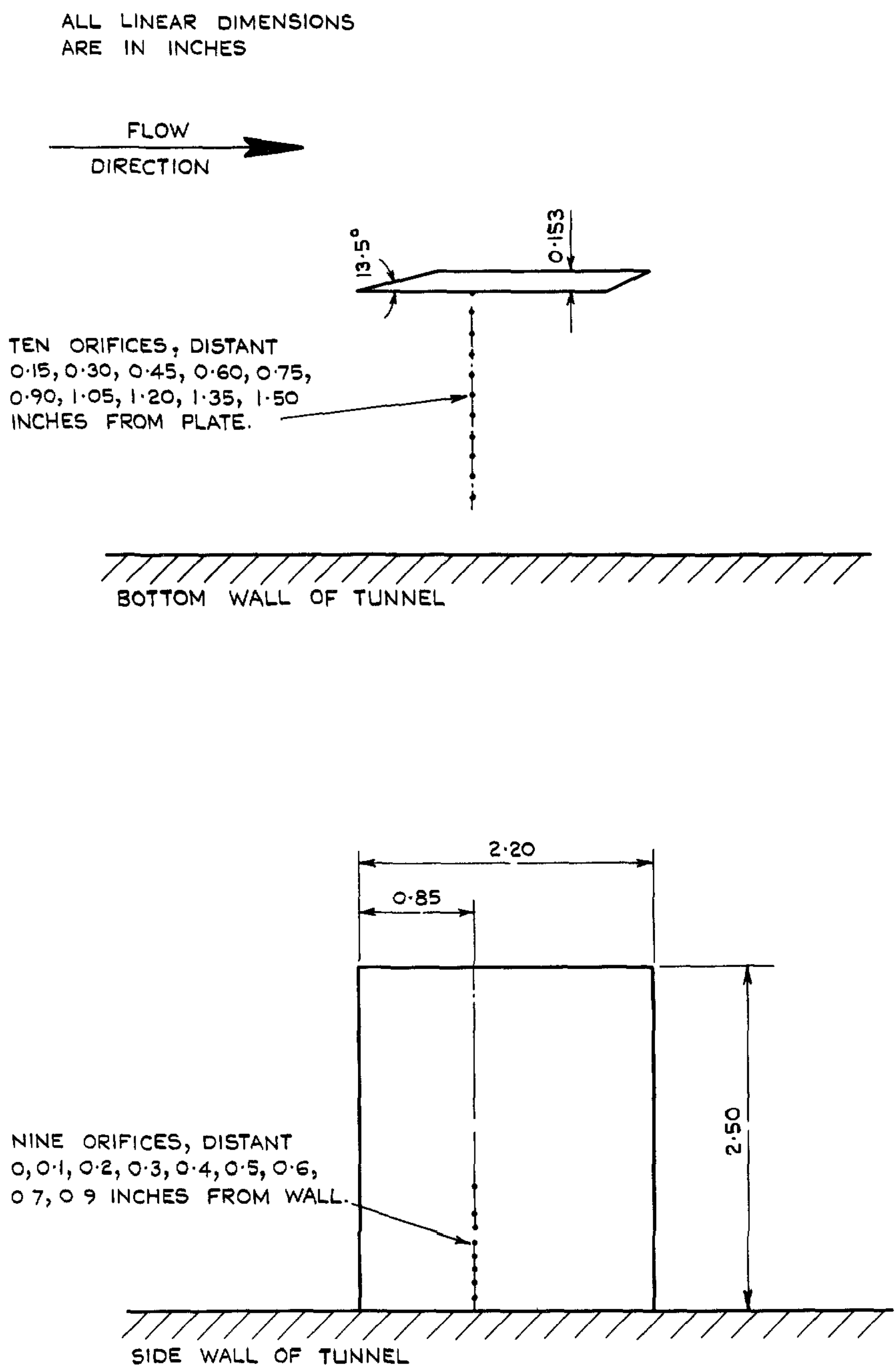


FIG. 1. SHOCK - PRODUCING PLATE.

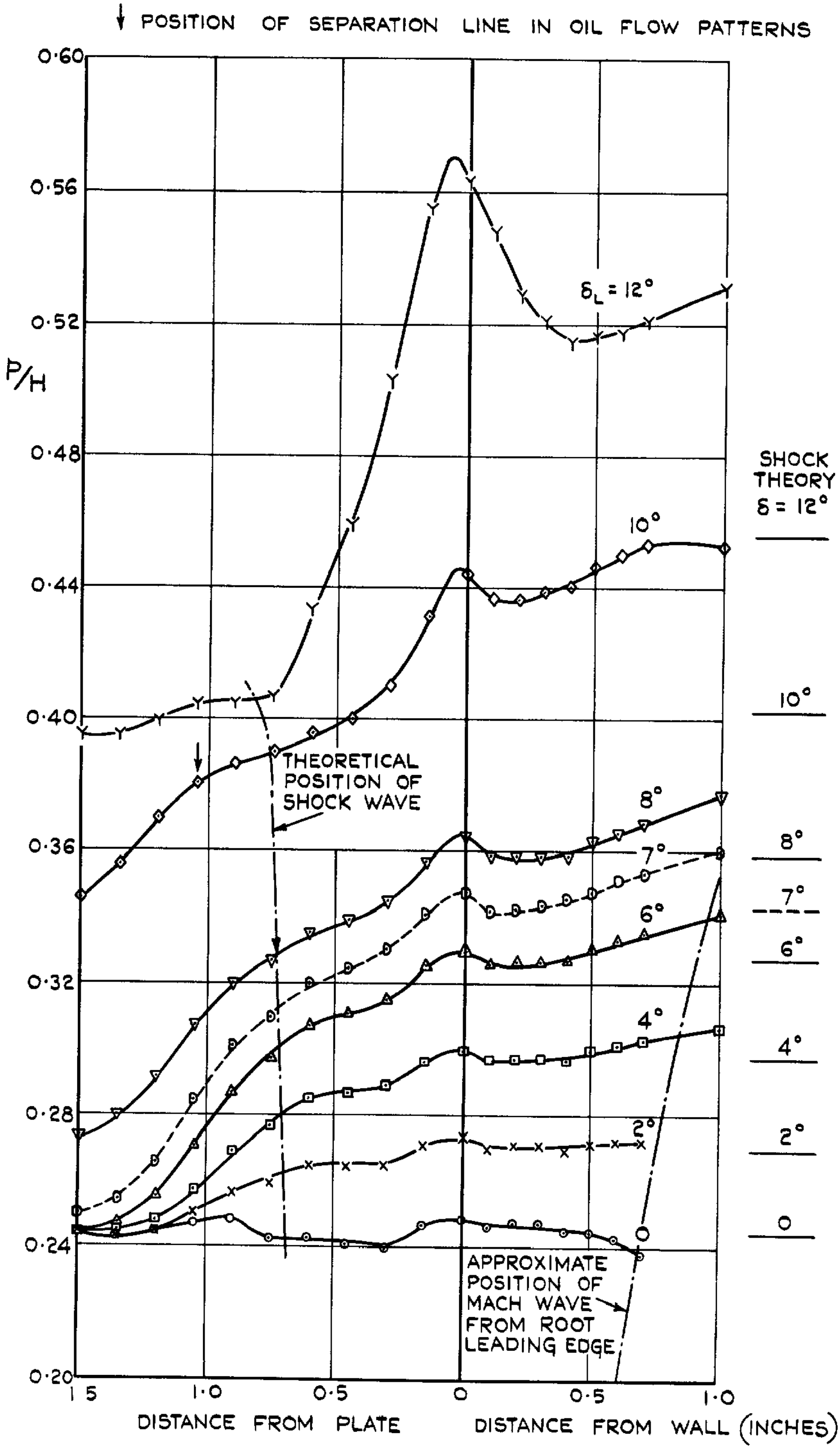
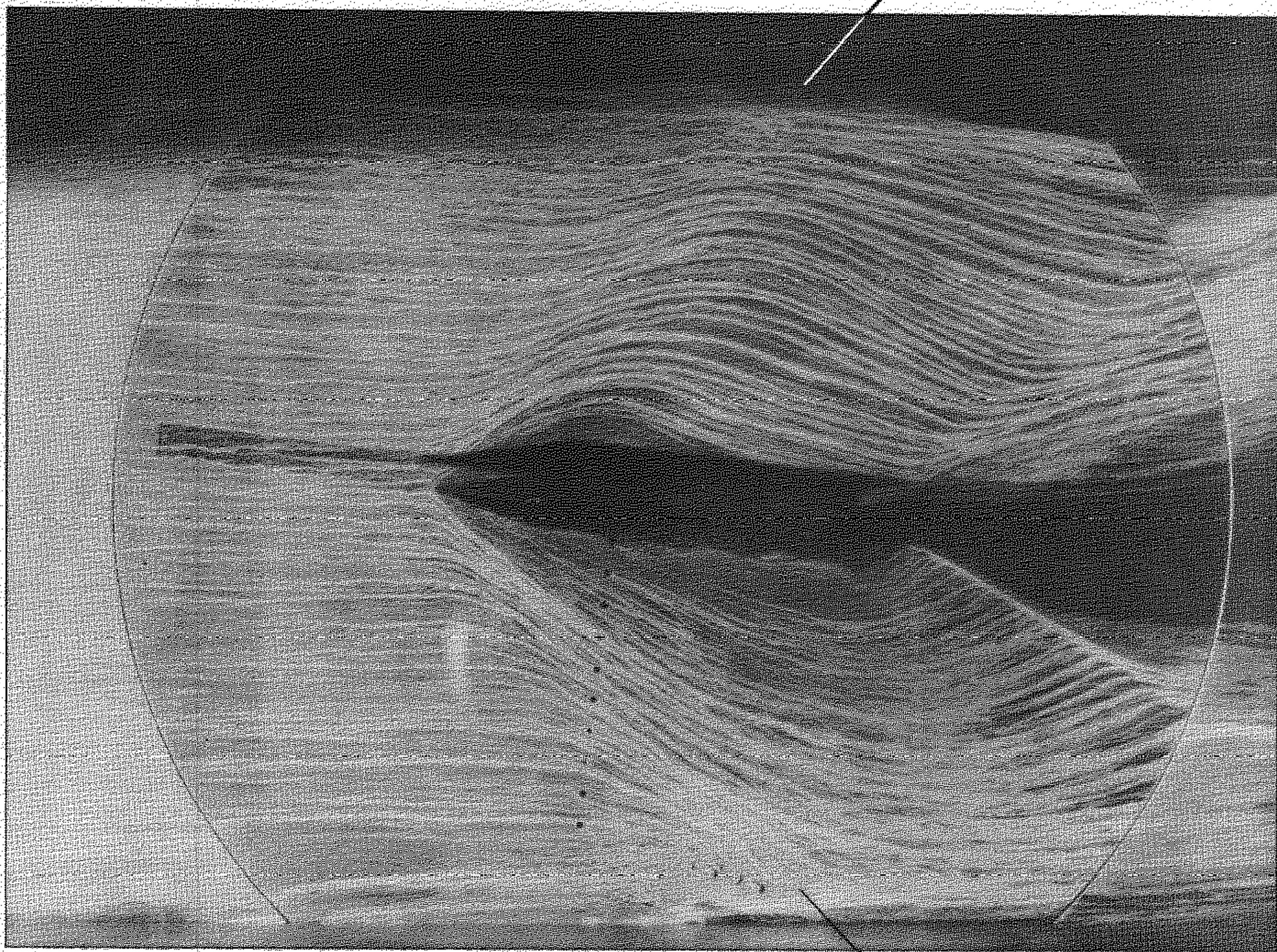
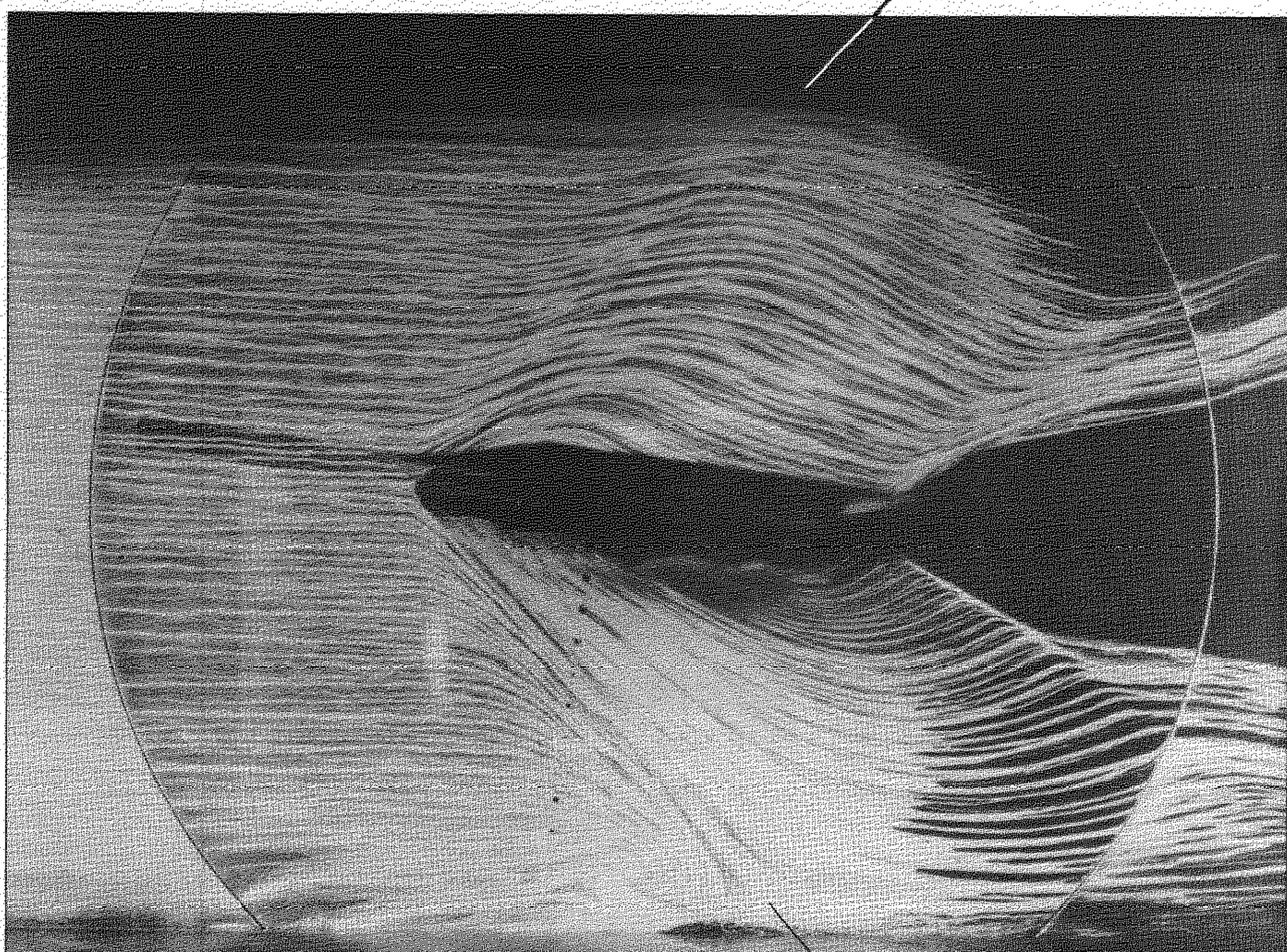


FIG. 2. PRESSURE DISTRIBUTIONS AT M = 1·6.



a. $\delta_L = 7^\circ$, $\delta_u = 6.5^\circ$



b. $\delta_L = 8^\circ$, $\delta_u = 5.5^\circ$

FIG.3 a & b. OIL FLOW PATTERNS AT M = 1·6

ANGULAR POSITION
OF SHOCK WAVE

c. $\delta_L = 10^\circ$, $\delta_u = 3.5^\circ$ SHOCK WAVE

(3rd. PHOTOGRAPH IN RUN)

FIG.3 c. OIL FLOW PATTERNS AT M = 1·6

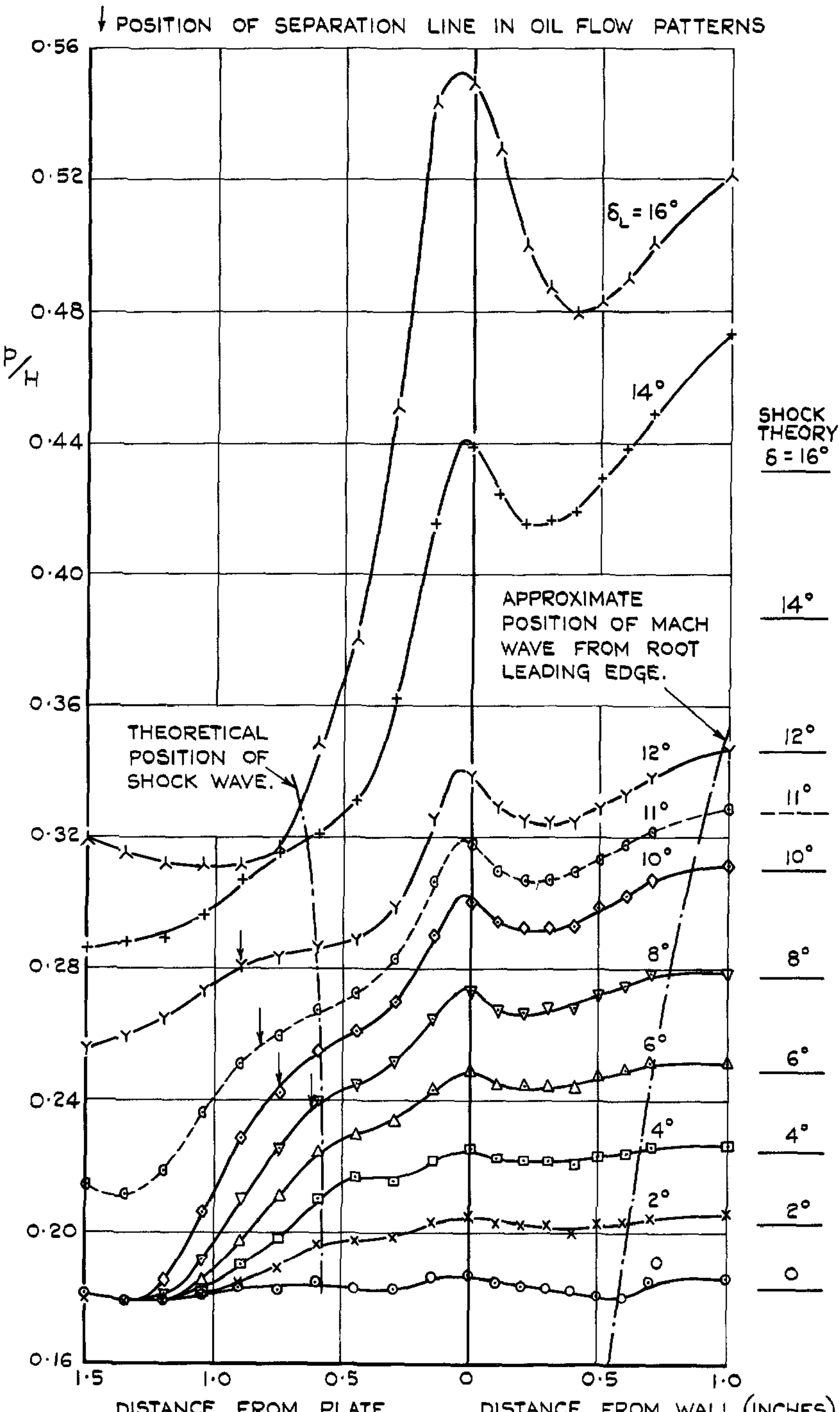
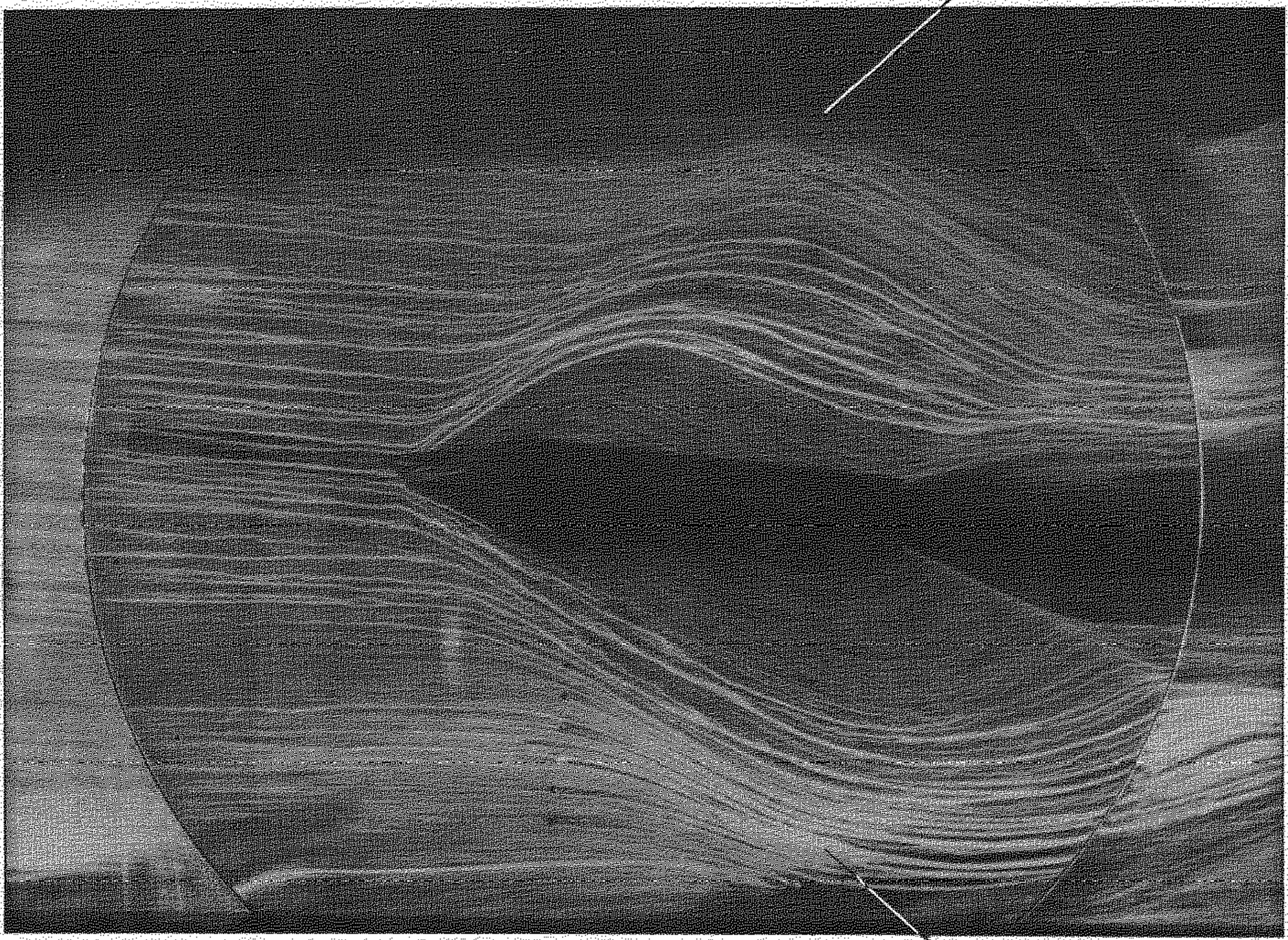
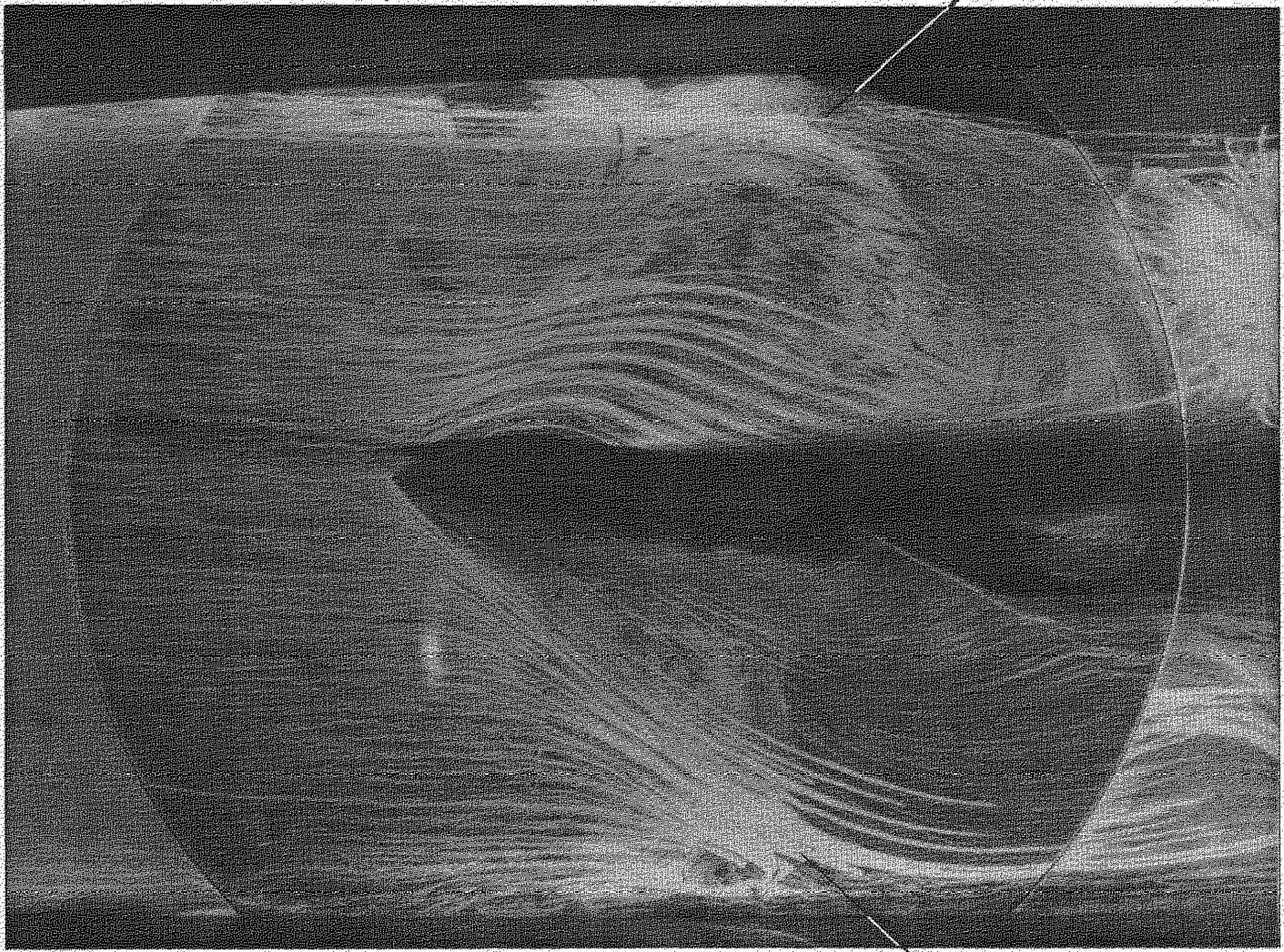


FIG.4. PRESSURE DISTRIBUTIONS AT M = 1·8.



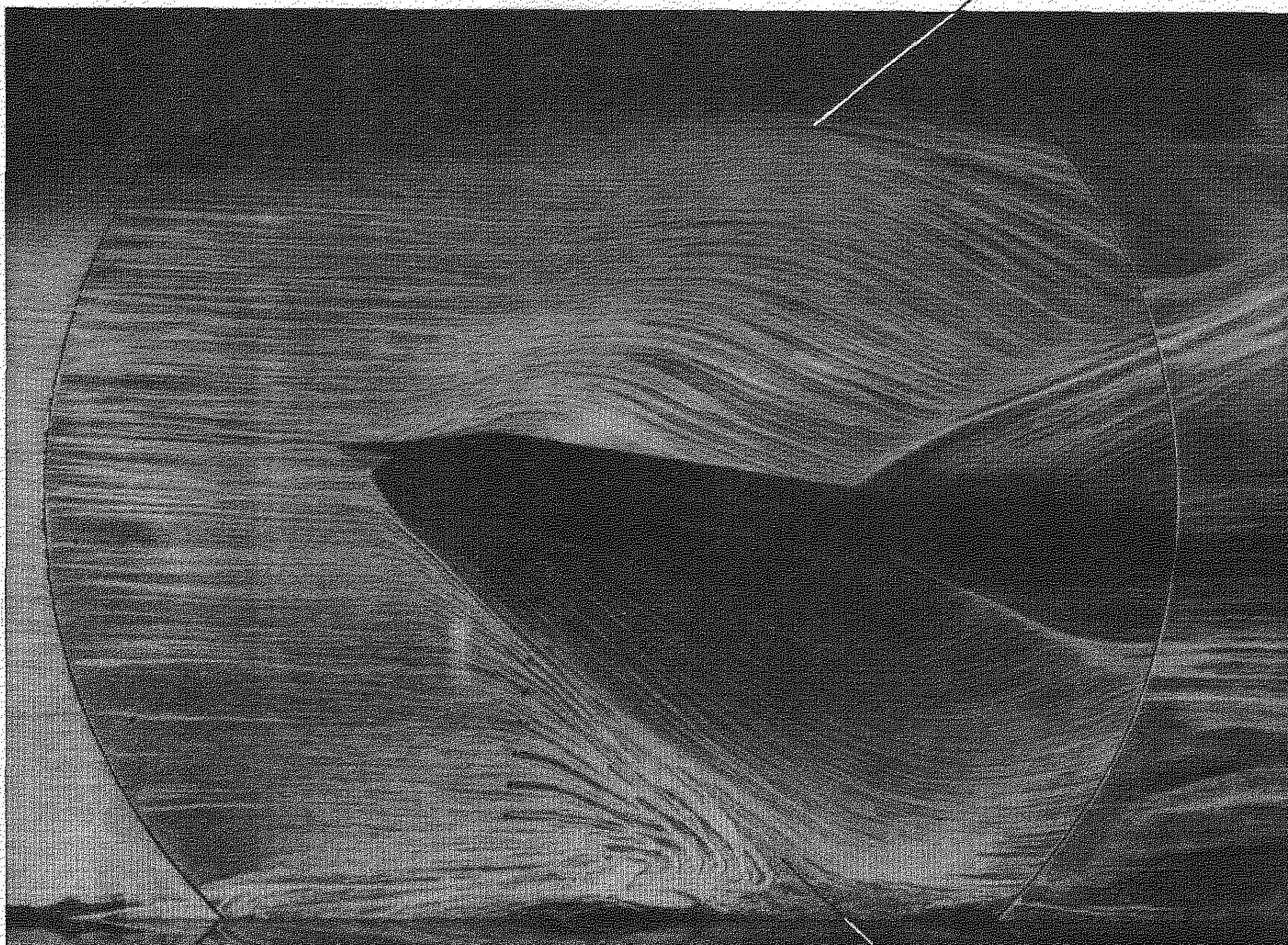
a. $\delta_L = 6^{\circ}$, $\delta_u = 7.5^{\circ}$



b. $\delta_L = 8^{\circ}$, $\delta_u = 5.5^{\circ}$

FIG.5 a & b. OIL FLOW PATTERNS AT $M = 1{\cdot}8$

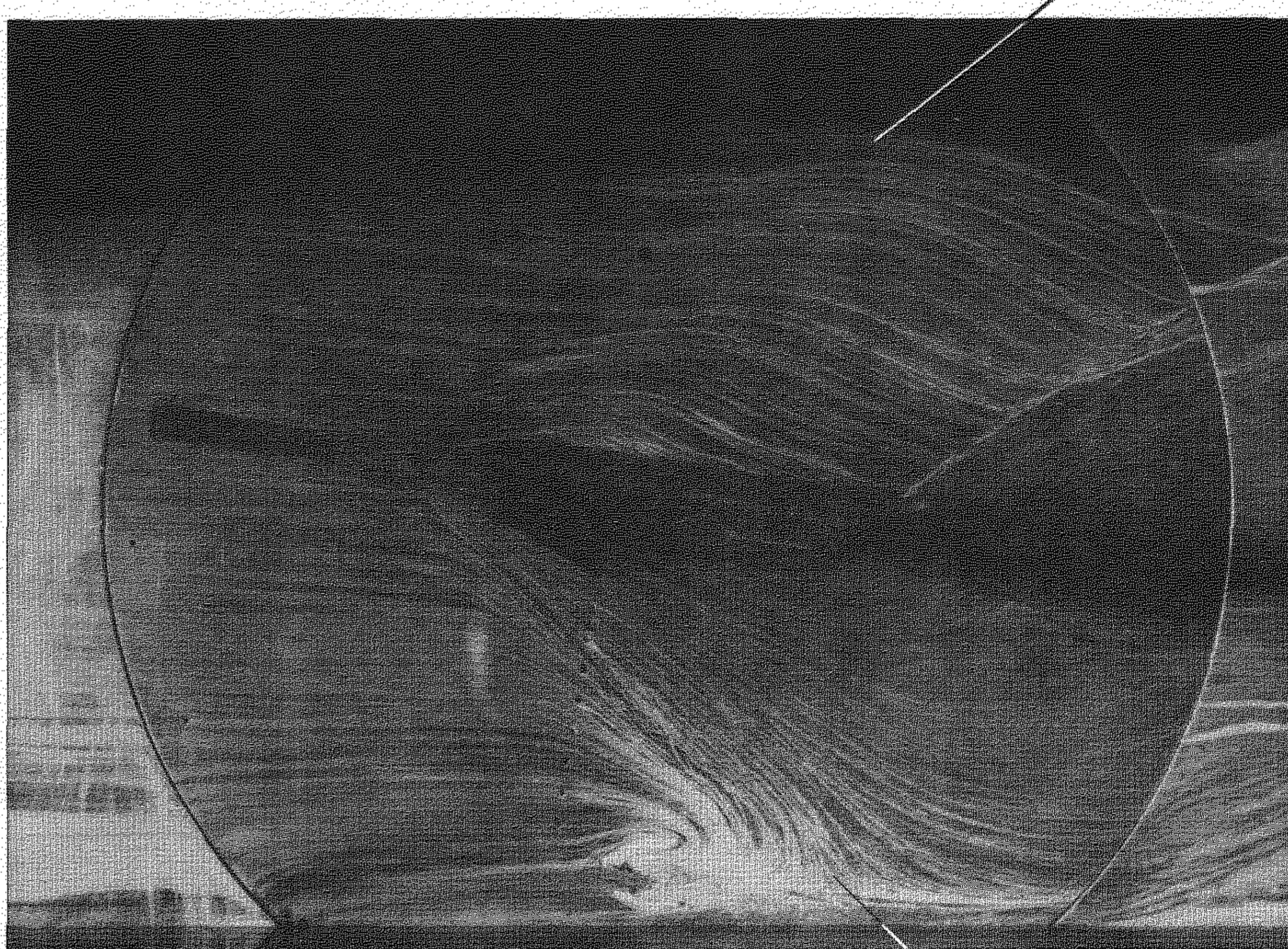
ANGULAR POSITION OF SHOCK WAVE



SHOCK WAVE

c. $\delta_L = 9^\circ$, $\delta_u = 4.5^\circ$

SHOCK WAVE



SHOCK WAVE

d. $\delta_L = 10^\circ$, $\delta_u = 3.5^\circ$

FIG.5 c & d. OIL FLOW PATTERNS AT M = 1·8

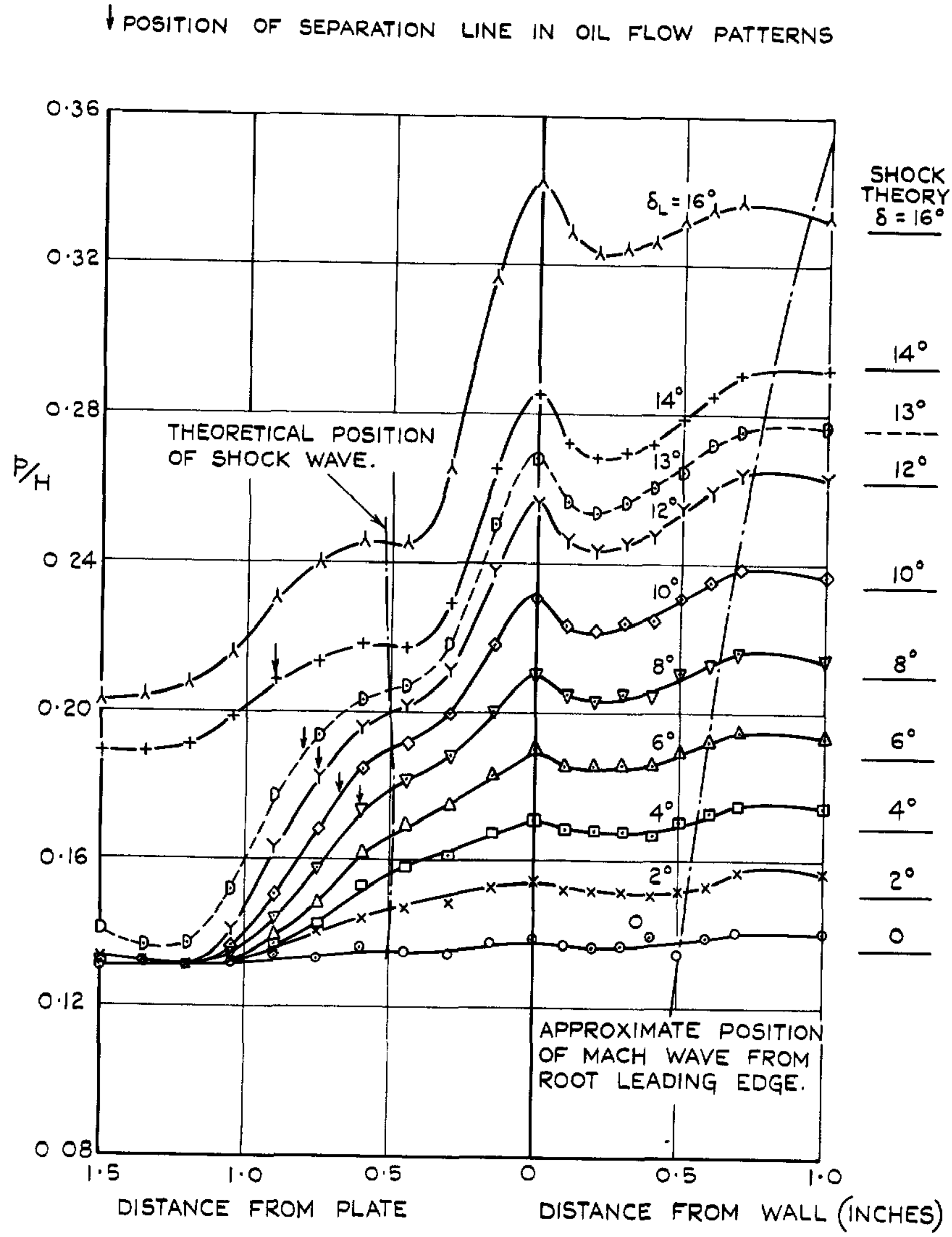
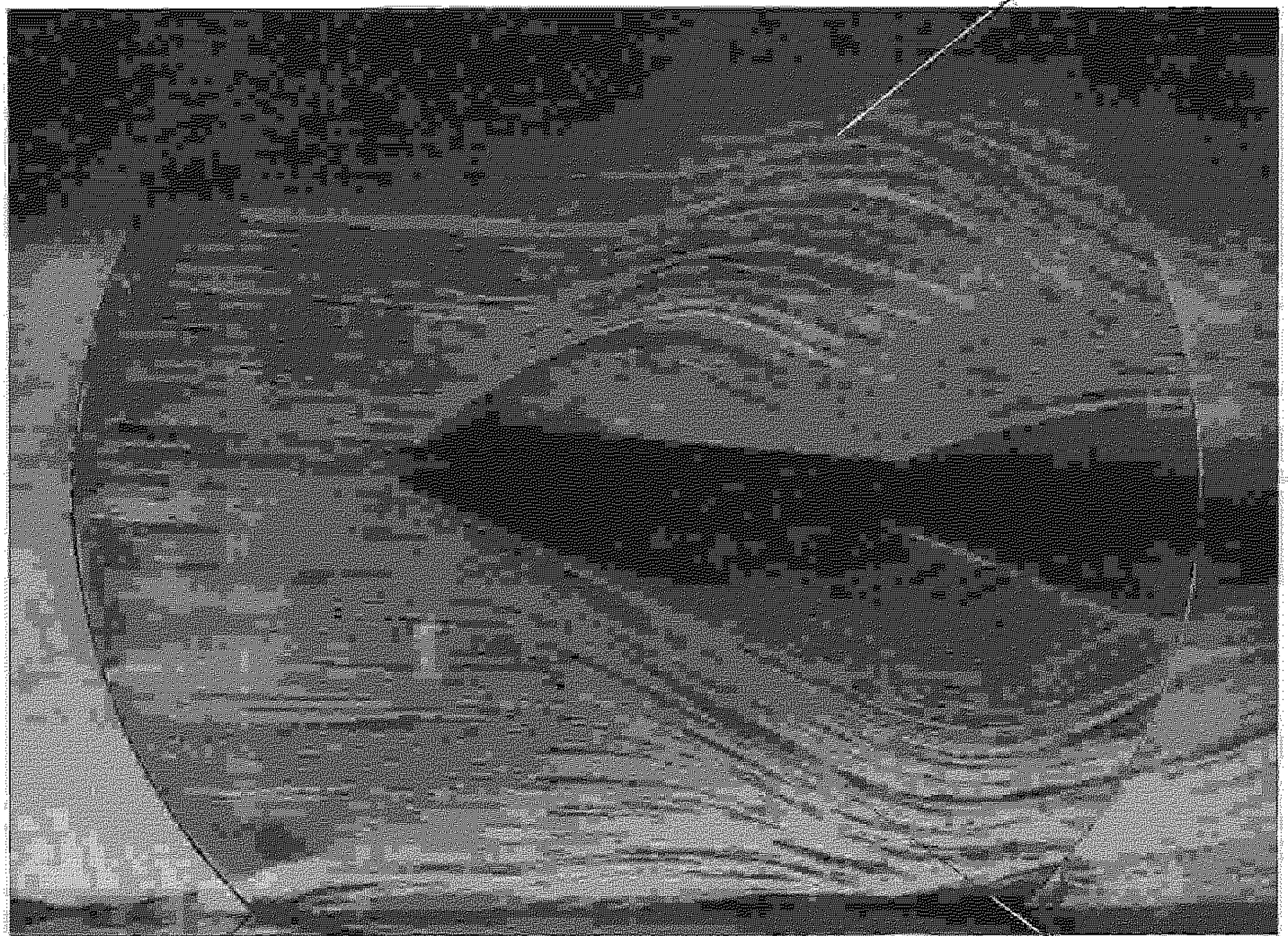
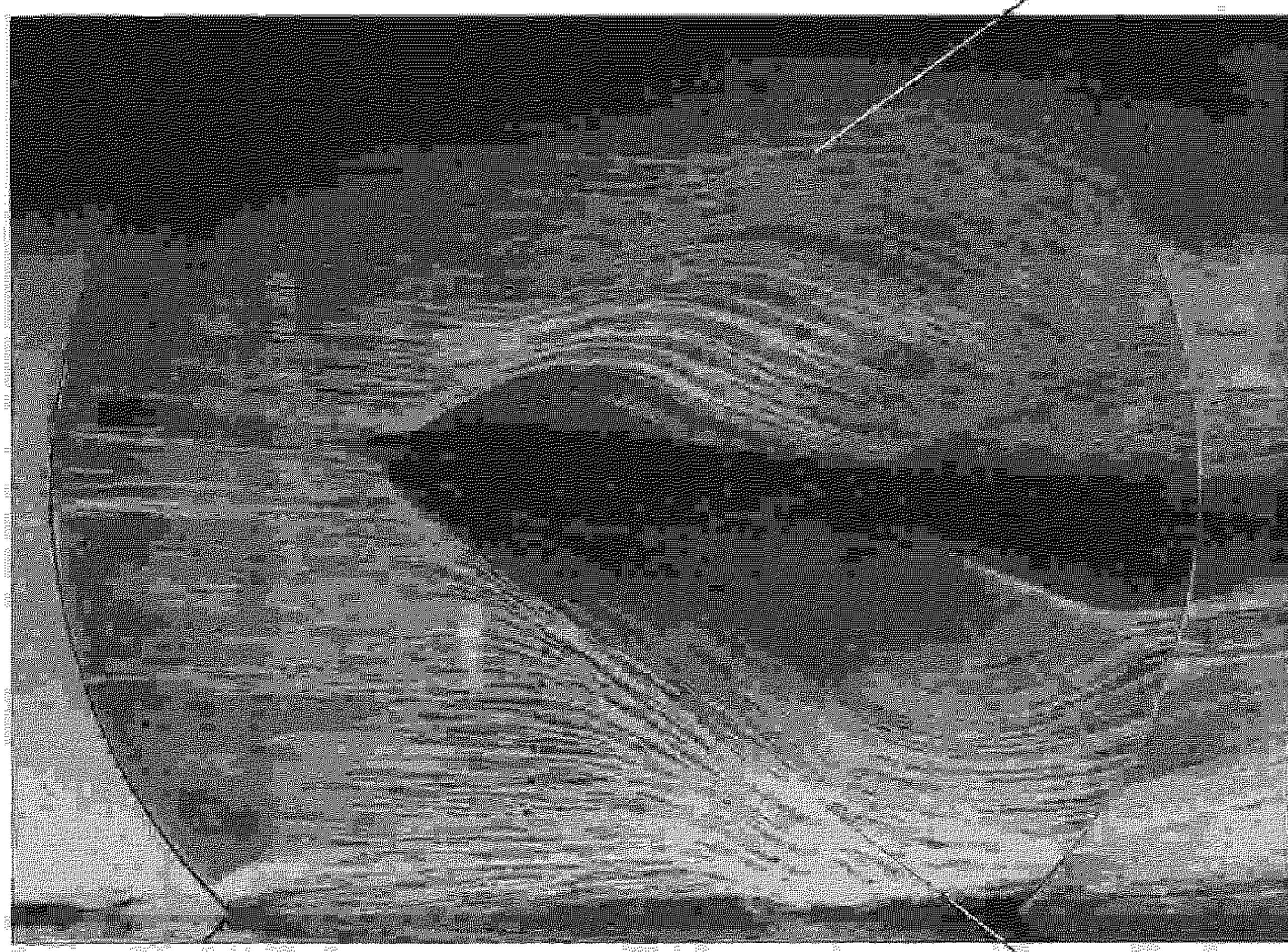


FIG. 6. PRESSURE DISTRIBUTIONS AT M = 2·0.

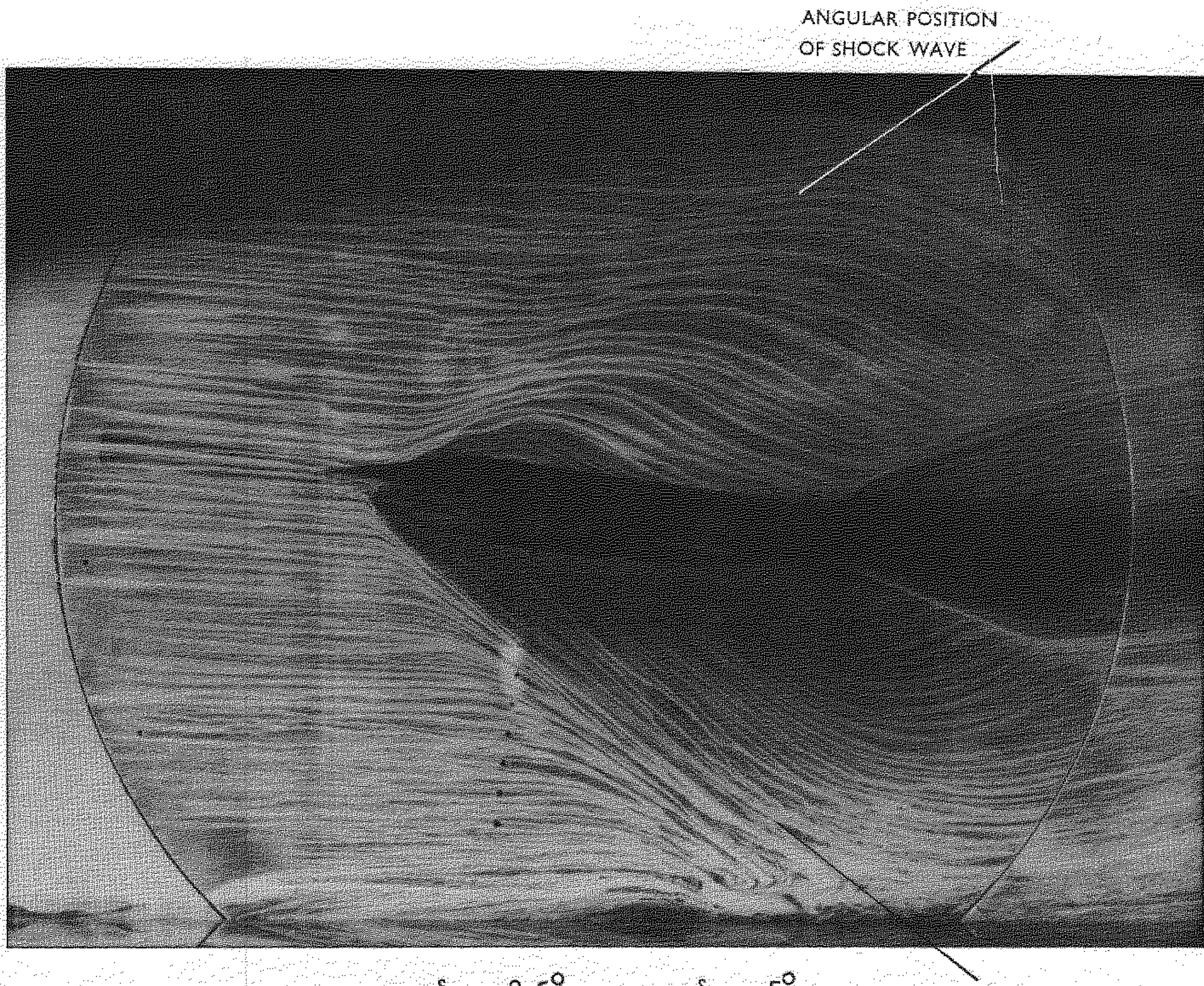


a. $\delta_L = 6^\circ$, $\delta_u = 7.5^\circ$

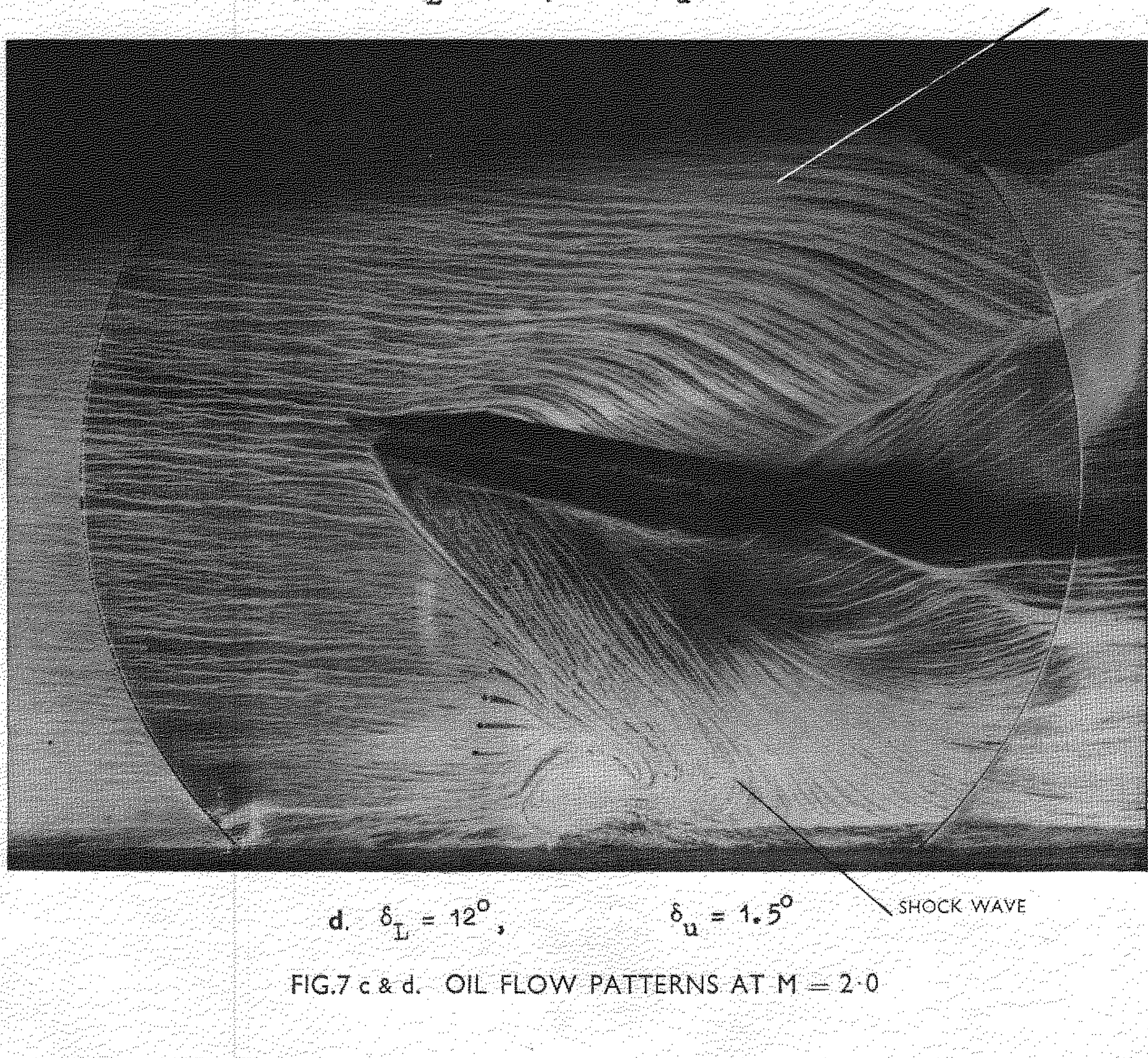


b. $\delta_L = 7.5^\circ$, $\delta_u = 6^\circ$

FIG.7 a & b. OIL FLOW PATTERNS AT M = 2·0



c. $\delta_L = 8.5^\circ, \qquad \delta_u = 5^\circ$



d. $\delta_L = 12^\circ, \qquad \delta_u = 1.5^\circ$

FIG.7 c & d. OIL FLOW PATTERNS AT M = 2·0

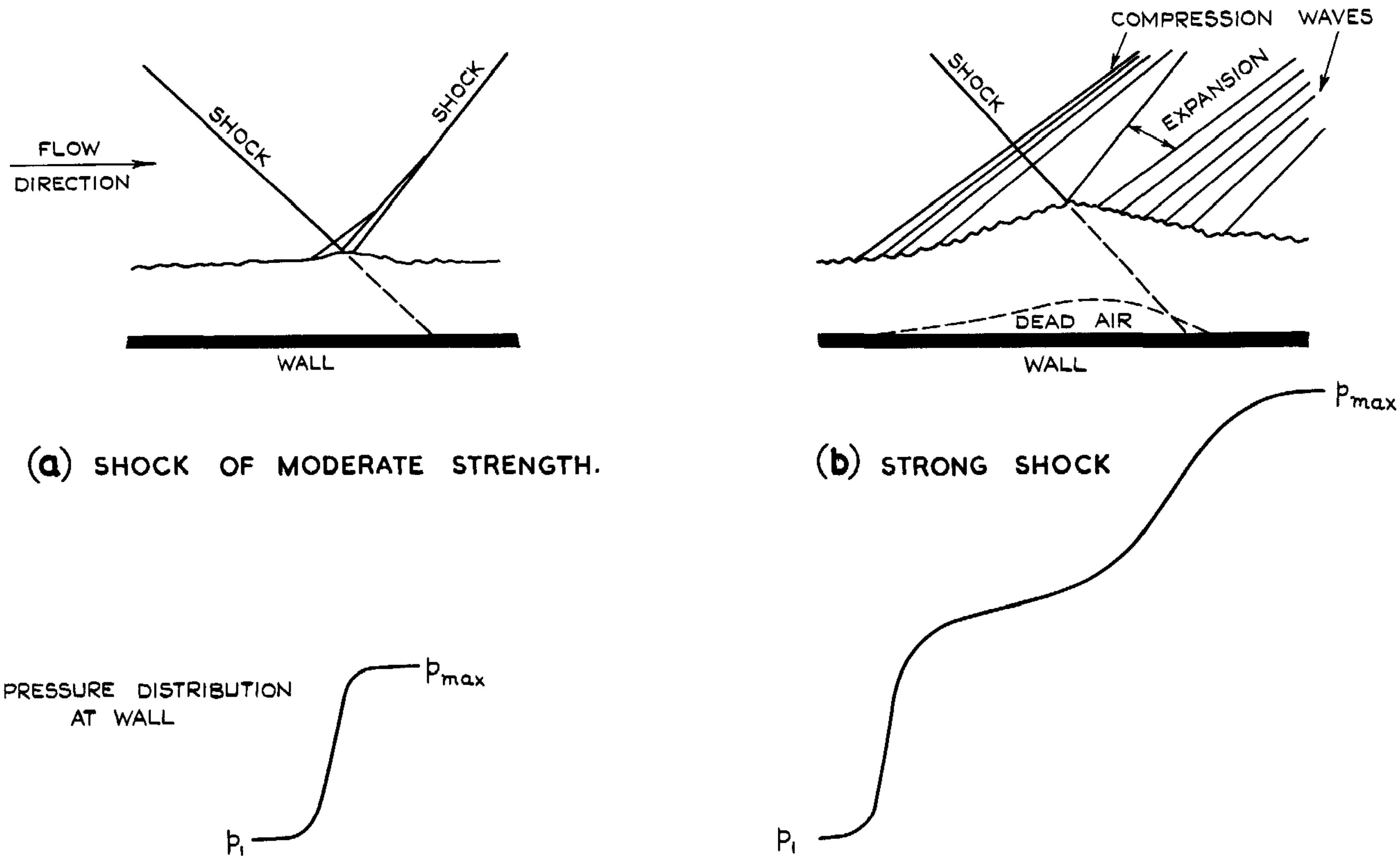


FIG. 8. TWO DIMENSIONAL INTERACTION BETWEEN A SHOCK WAVE AND A TURBULENT BOUNDARY LAYER (AFTER HOLDER, ET ALIA.)

	SEPARATED	ATTACHED
UPPER SURFACE	▲	△
LOWER SURFACE	▼	▽

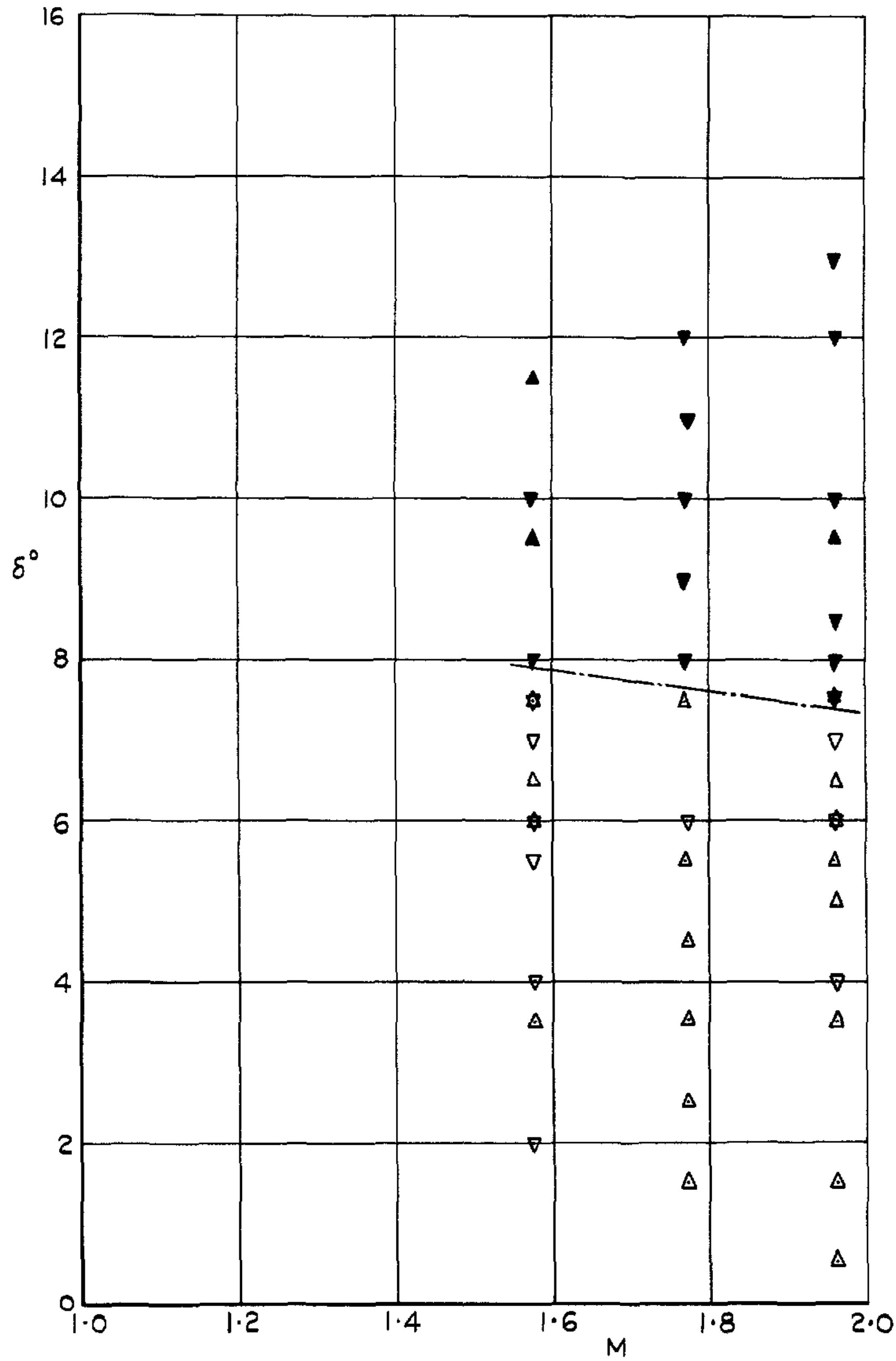


FIG.9. FLOW DEFLECTION THROUGH A SHOCK SUFFICIENT TO SEPARATE A TURBULENT BOUNDARY LAYER.

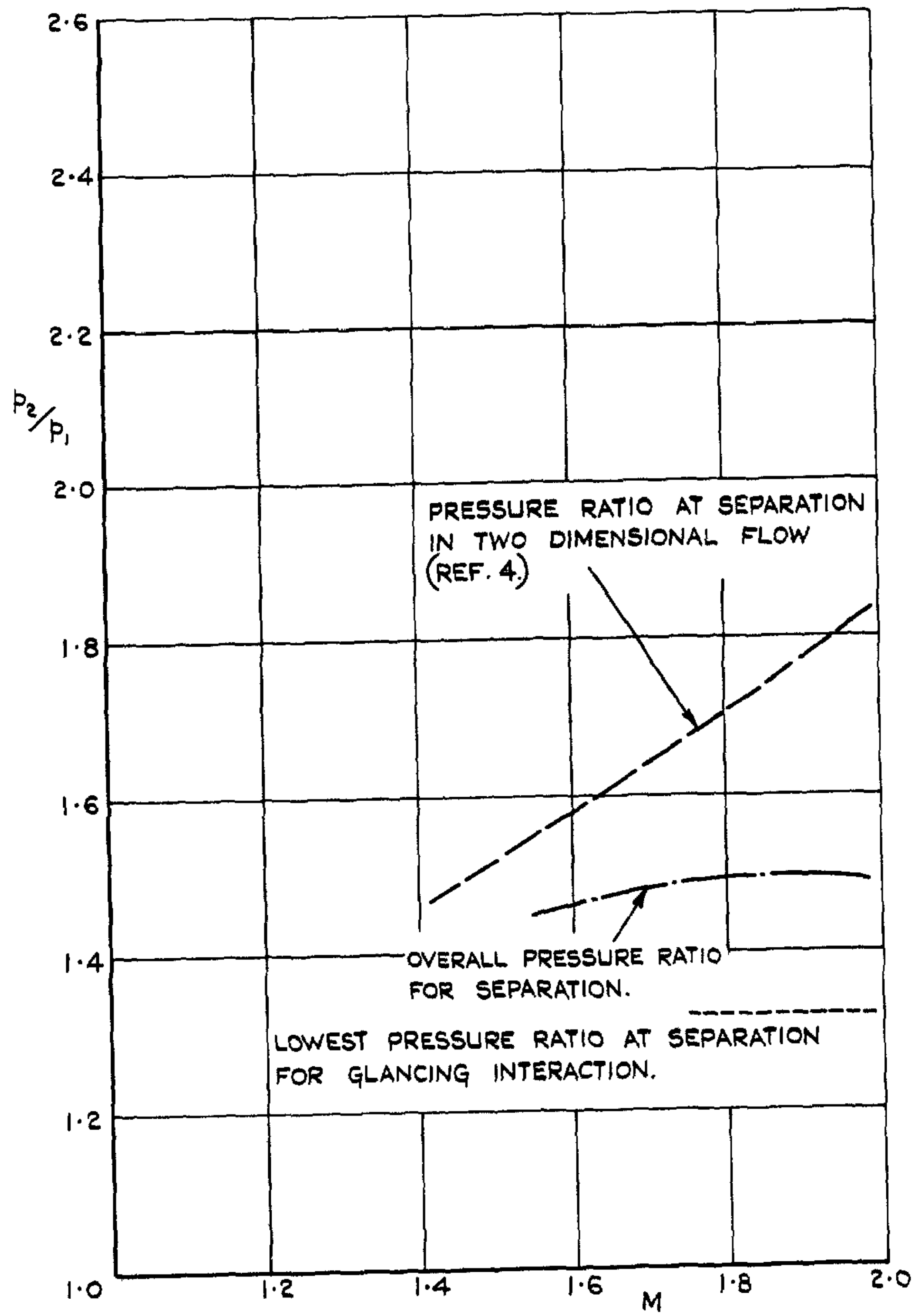
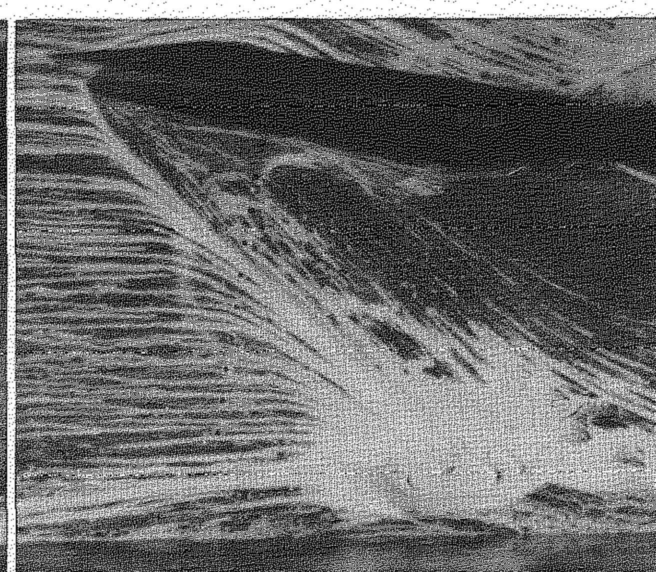


FIG. 10. SHOCK STRENGTH SUFFICIENT TO SEPARATE A TURBULENT BOUNDARY LAYER.

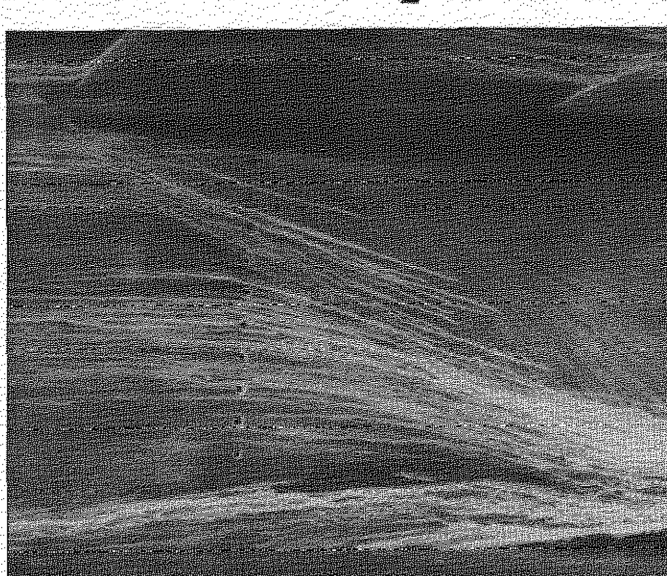
FOR 1st. PATTERN IN RUN
SEE FIG.7 d, $\delta_L = 12^\circ$

2nd. PATTERN IN RUN, $\delta_L = 14^\circ$

3rd. PATTERN IN RUN, $\delta_L = 10^\circ$

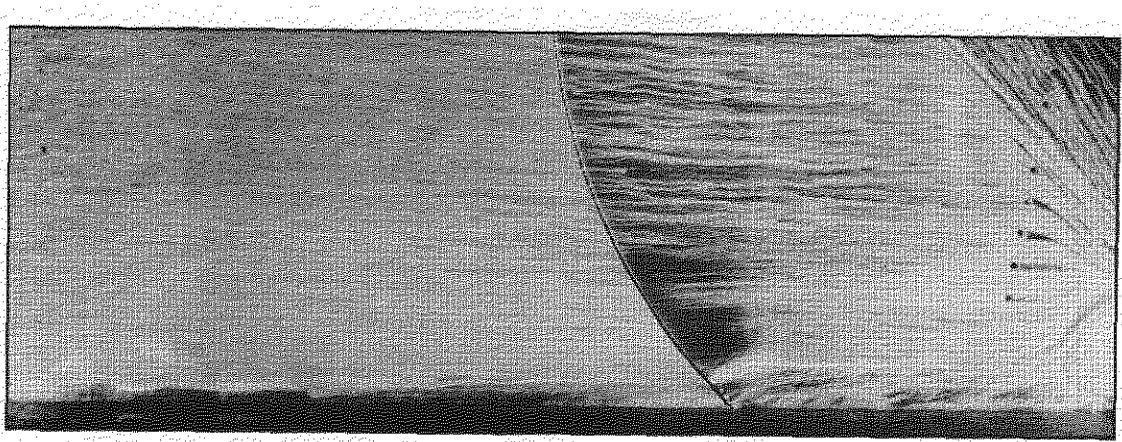


1st. PATTERN IN SEPARATE RUN, $\delta_L = 10^\circ$

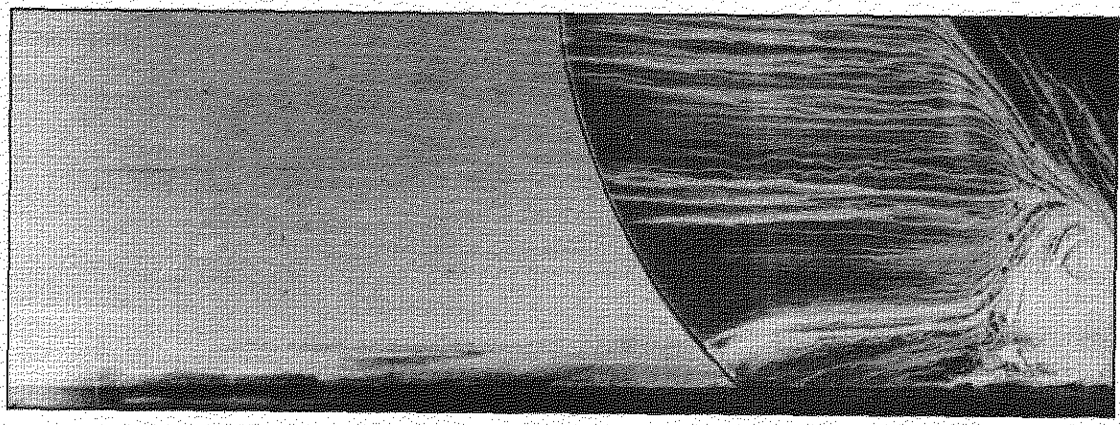


4th. PATTERN IN RUN, $\delta_L = 4^\circ$

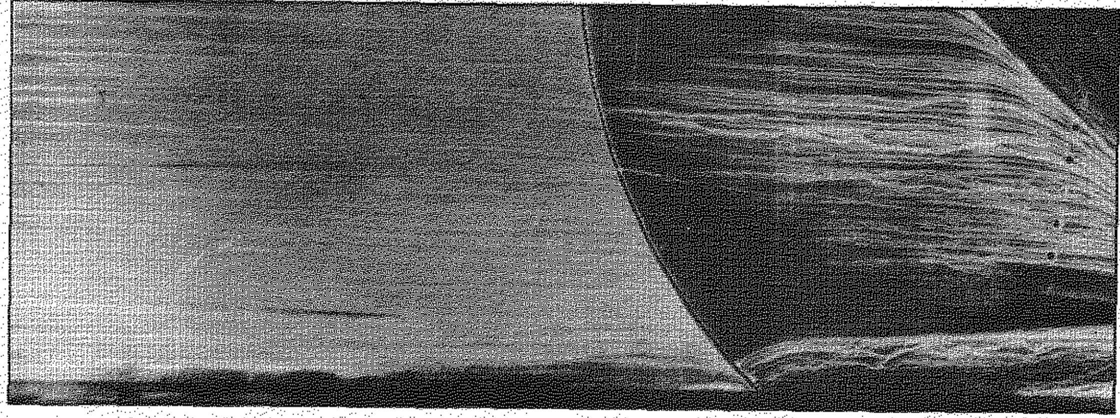
FIG.11. OIL FLOW PATTERNS PRODUCED CONSECUTIVELY IN ONE RUN AT $M = 2{\cdot}0$



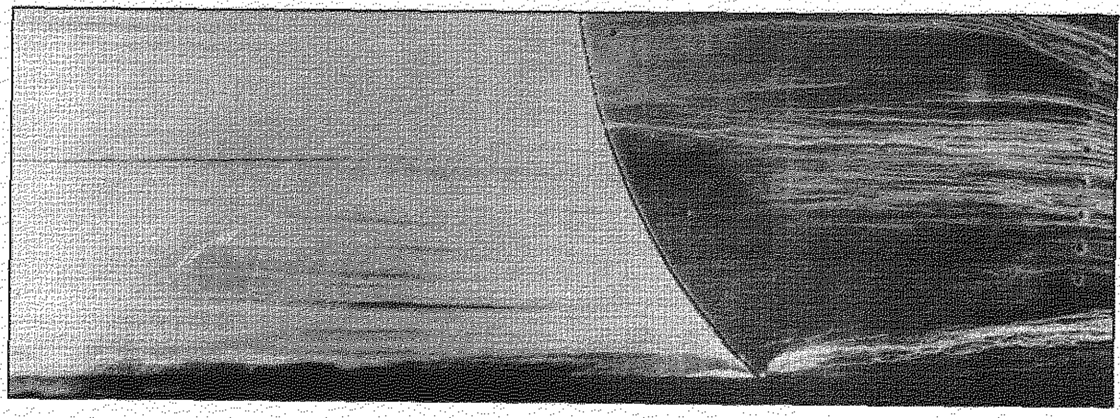
APPROXIMATELY 6 MINUTES FROM START OF RUN



APPROXIMATELY 12 MINUTES FROM START OF RUN



APPROXIMATELY 18 MINUTES FROM START OF RUN



APPROXIMATELY 24 MINUTES FROM START OF RUN

FIG. 12. DEVELOPMENT OF STRAIGHT OIL FLOW LINES WITH TIME.

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Stanbrook, A. July, 1960.

An experimental study has been made at Mach numbers from 1.6 to 2.0 of the interaction between the turbulent boundary layer on a side wall of a wind tunnel and the shock wave produced by a plate mounted on the wall. Under these conditions the shock wave/boundary layer interaction was three dimensional at least over the region investigated (up to 10 boundary layer thicknesses from the plate). It was found that the boundary layer was separated by a shock wave of strength $p_2/p_1 = 1.5$, i.e. a flow deflection between 7.5^{0} and 8^{0}. Interactions of this type occur on the sides of fuselages at the wing/fuselage junction and may therefore be important with regard to the design of waisted shapes.

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