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A Study at a Mach Number of 2.01 of the Shock Boundary-Layer Interaction Resulting from the Deflection of a Wedge Mounted on a Bypass Plate

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A STUDY AT A MACH NUMBER OF 2.01 OF THE SHOCK BOUNDARY-LAYER INTERACTION RESULTING FROM THE DEFLECTION OF A WEDGE MOUNTED ON A BYPASS PLATE

A Thesis

Presented to

The Faculty of the Department of Physics

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In Partial Fulfillment

Of the Requirements for the Degree of

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APPROVAL SHEET

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ABSTRACT

Oil-flow photographs and shadowgraphs were obtained of the flow on the surface on a bypass plate in order to study the interaction of a shock wave with a turbulent boundary layer. The shock wave was generated by a wedge mounted at various deflection angles on the bypass plate.

Deflection angle at which the boundary layer on the plate separated from the surface was found to vary depending upon the criteria used to define the angle.

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INTRODUCTION

A shock wave forms ahead of any body in supersonic flight because of the finite compressive disturbances created at the nose of the body by its motion through the air. This shock remains fixed relative to the body if the velocity is constant. It stands ahead of blunt shapes but may be attached to pointed shapes. For a wedge with a 10° apex angle in a supersonic stream at a Mach number of 2.01 the shock wave is attached to the leading edge. This shock is straight, and behind it the flow consists of uniform streams parallel to the wedge faces. Since the flows above and below the wedge are independent, the flow over the surfaces of inclined wedges can be considered separately.

When the lower surface of the wedge is aligned with the flow direction, the flow on this side of the wedge is undisturbed and there is no shock. As the lower surface is inclined to the flow a shock wave is formed at the leading edge. The compression causing the shock wave to form becomes greater as the deflection angle is increased so that the shock intensity or strength is increased.

For this 10° wedge, when the deflection angle of the lower surface (δ_L, shown in fig. 1) is less than 10° there will be a shock formed on the upper surface which will decrease in intensity as δ_L increases. (δ_L is used as a reference only because of convenience.) At $\delta_L = 10^{\circ}$ the upper surface will be aligned with the flow and there is no disturbance. For values of δ_L greater than 10° the flow expands around the leading edge of the wedge onto the upper surface.

When air is flowing with a given velocity over a surface, the air because of its viscosity tends to adhere to the surface. This means that frictional forces retard the motion of the air in a thin layer near the surface. This layer is called the boundary layer and the velocity of the air increases from zero at the surface to a value which corresponds to the external frictionless flow. In some cases the thickness of boundary layer increases considerably in the downstream direction, and the flow in the boundary layer becomes reversed. This causes the decelerated particles of air to be forced outside the boundary layer so that the boundary layer is separated from the surface, or in other words boundary-layer separation has occurred. The flow in the boundary layer may be either laminar or turbulent. In a laminar boundary layer the air moves smoothly in layers or lamina which slip over one another while in a turbulent boundary layer the flow has an irregular, eddying or fluctuating nature.

Many situations arise in which the interaction of boundary layers with shock waves is of practical importance. Such interactions occur at transonic and supersonic speeds over wing surfaces, at the juncture of the wing and fuselage, near deflected controls and in many other cases.

The interaction of a shock wave and a turbulent boundary layer may be divided into two parts: (1) the case where the change in flow direction through the shock wave is in a plane normal to the surface on which the boundary layer is being studied; and (2) the case where the change in flow direction through the shock is in a plane parallel to surface under study.

The problems associated with the first case have been the subject of numerous investigations. Fage and Sargent[1] examined the interaction of a normal shock wave with the turbulent boundary layer on the flat wall of a supersonic tunnel. Kepler and Bogdonoff[2] studied the separation of the turbulent boundary layer and the associated shock-wave pattern caused by the flow over a two-dimensional step. Gadd, Holder, and Regan[3] investigated the interaction between the boundary layer on a flat plate and a shock wave produced either externally, by a wedge in the supersonic mainstream, or from within the boundary layer, by a wedge held in contact with the plate. Gadd and Holder[4] reviewed some of the more recent work in these areas. In general, these and other investigations have shown that the flow is very dependent upon whether the boundary layer is laminar or turbulent and, if laminar initially, whether or not transition to turbulent flow occurs within the region of interaction. The separation of the boundary layer from the surface ahead of the shock, the conditions under which this separation occurs, and the behavior of the separated boundary layer were found to be important in explaining the differences between the interactions observed with laminar and turbulent boundary layers. Reshotko and Tucker[5] have shown theoretically, and verified with available experimental data, that the pressure rise across a shock is a significant factor in the separation of a turbulent boundary layer. This pressure rise is a function of the local Mach number outside the boundary layer and ahead of the shock.

The problems arising from case two have been less extensively studied. This type of interaction is called "glancing interaction" by Stanbrook[6] who has studied the phenomena to provide information on the pressure rise across a shock sufficient to cause the boundary layer to separate from the surface and to provide information on the type of flow which occurs under these conditions.

The object of this investigation was to obtain oil-flow photographs and shadowgraphs of the flow along the surface of a plate immersed in flow fields having shocks of known intensities in order to study the interaction of the shock wave with a turbulent boundary layer. A wedge mounted at various deflection angles on a bypass plate was used to generate the flow fields. Deflection angle varied from 0° to 20° in 5° increments. The interaction of the shock wave from the wedge with the turbulent boundary layer on the bypass plate is the same type (glancing interaction) as that studied by Stanbrook[6]. Deflection angle for boundary-layer separation was found to depend upon the criteria used to define the angle. One of the values obtained for the deflection angle for separation indicated that the phenomena associated with the two types of shock wave turbulent boundary-layer interaction may be the same.

CHAPTER I

EXPERIMENTAL PROCEDURE

Description of test setup.- This investigation was conducted at a Mach number of 2.01 in the Langley 4- by 4-foot supersonic tunnel which has provisions for the control of the pressure, temperature, and humidity of the enclosed air. The reference pressure for calculations is the tunnel stagnation pressure (the pressure measured at a location in the tunnel where the velocity of the enclosed air is zero).

Reynolds number, which is directly proportional to density, velocity, and length and inversely proportional to the viscosity of the air, is usually based on some characteristic length of the model being tested. In this investigation there is no particular characteristic length upon which to base Reynolds number so Reynolds number per unit length is used.

A wedge with a 10° apex angle was mounted on the boundary-layer bypass plate which is located about 10 inches from the tunnel side wall. A strip of carborundum grains was placed parallel to the leading edge of the bypass plate to insure that the boundary layer on the plate was turbulent. Figure 1 shows a schematic drawing of the test setup. Oil-flow photographs and shadowgraphs were obtained for wedge deflection angles of 0° to 20° at tunnel stagnation pressures of 1200 and 2400 pounds per square foot absolute. These pressures correspond to Reynolds numbers per foot of 2×10^6 and 4×10^6, respectively. The camera was located outside the tunnel, below and ahead of the model, in order to obtain the shadowgraphs. Both types of photographs were obtained at the same camera position with a lens aperture of f-16. Exposure time was 6 seconds for the oil-flow photographs and 1/2 second for the shadowgraphs using Kodak Tri-X pan film.

Oil film technique.- The oil film technique used during this investigation consists of coating a model surface with a fluorescent oil and observing the oil under ultraviolet light. During a test, the airflow sweeps the oil along the surface, so that the oil develops a pattern of striations indicative of the flow conditions on the surface. Generally, several observations may be made during the course of a test.

Various types of oil may be used depending upon the operating conditions of the wind tunnel. For the Langley 4- by 4-foot supersonic tunnel, a mixture of three parts of Navy gear oil No. 6135 and two parts kerosene has been found to be the best mixture for tunnel stagnation pressures of about 10 pounds per square inch absolute. At higher pressures a thicker mixture is needed so less kerosene is used. Similarly, at lower pressures more kerosene is used to obtain a thinner mixture. Approximately 1 cubic centimeter of fluorescent dye per liter of oil was added to supplement the natural fluorescence of the oil and kerosene mixture.

A good source of ultraviolet light is a mercury vapor lamp with an ultraviolet filter. For each square foot of model surface area, two 100-watt EH4 mercury vapor lamps with ultraviolet filters will provide sufficient illumination to photograph the flow when placed 30 inches from the model. More or less light may be desired for a particular test setup and may be obtained by adding lamps or varying the distance from the light source to the model.

With presently available high-speed films and with proper use of the ultraviolet light source any camera will produce satisfactory results. A filter should be placed over the camera lens to absorb ultraviolet and visible blue light that might reach the camera from the ultraviolet lamp or by reflections. The Kodak Wratten filter numbers 2A or 2B will serve this purpose.

Oil-flow photographs are shown in figure 2. Figure 2(e) is a typical example of the pattern of striations formed by the airflow sweeping the oil along the surface. Since the camera is below and ahead of the wedge the root section appears further forward than the tip section. The large circle visible in the photograph is the turntable on which the wedge is mounted so that deflection angle may be changed. The smaller circles visible at other locations are bolt heads from mounting the bypass plate on the tunnel sidewall. The turntable and bolt heads are flush with the surface of the plate. Since the flow over upper surface of the wedge is not part of this investigation only the pattern formed below the wedge in figure 2(e) will be described. Ahead of the leading edge of the root section the oil has formed a pattern of lines which are parallel to the direction of the airflow. When these lines reach the disturbance caused by the wedge deflection, they are turned away from the wedge. This disturbance at the leading edge affects the streamlines for some distance below the wedge. Behind this disturbance the oil streamlines near the wedge show the influence of the flow expansion which occurs at the trailing edge. The bending of the oil streamlines behind the wedge is indicative of this influence. Further away from the wedge the oil lines are turned toward the line formed by the disturbance at the leading edge.

Shadowgraph technique.- The shadowgraph technique is a convenient and simple method of making shock waves visible. Basically the method depends on the fact that light passing through a density gradient in a gas (and therefore through a gradient in the index of refraction) is deflected in the same way as though it were passing through a prism. Parallel light from a small intense source is allowed to pass through the subject and fall directly on a screen. At the screen the intensity of the light is a function of the density variation in the gas through which the light has passed. When there is no flow, or when the density variation is constant after flow has been established, there will be no change in illumination on the screen because each light ray is deflected by the same amount. When there is a positive variation in the density gradient the light ray diverges and light intensity on the screen is decreased. Conversely, when the variation in the density gradient is negative the light rays converge and the intensity is increased. Sharp shadow images will be produced by rapidly varying density gradients as through a shock wave.

In this investigation the light source was an AH6 mercury vapor lamp emitting continuous light. The shadow image cast upon the bypass plate was photographed from a position ahead and below the model.

Figure 3(d) is typical of the shadowgraphs shown in figure 3. The shock is indicated by an arrow. Because of the latent fluorescence of the oil the oil-flow pattern is still visible. The vertical bar which partially obscures the shock is the shadow of the vertical support in the tunnel window. Shadows of various parts of equipment are also visible including, near the rear of the wedge, the ultraviolet lamps used for the oil-flow photographs.

CHAPTER II

EXPERIMENTAL RESULTS

The results of this investigation are concerned with the interaction between the shock wave formed by the lower surface of a wedge having a 10° apex angle and the boundary layer on the bypass plate on which the wedge is mounted or "glancing interaction".

Oil-flow photographs are presented in figure 2 for the various wedge deflections at a Reynolds number per foot of 2×10^{6}. The deflections produced by the lower surface provide information for deflection angles (δ_L) from 0° through 20°. In the photographs the airflow is from left to right. As previously stated the camera is below and ahead of the wedge so that the root section appears further forward than the tip section. At $\delta_L = 0^{\circ}$ (fig. 2(a)) especially, one must be careful not to take the dark triangle which is the tip section as the junction of the wedge root section and the bypass plate. As δ_L increases this junction of the wedge and the bypass plate is easier to locate.

Since there is no disturbance at the wedge leading edge at $\delta_L = 0^{\circ}$ (fig. 2(a)) the flow on the surface of the bypass plate is parallel to the wedge surface until the trailing edge where the flow expands around the corner of the wedge. At $\delta_L = 5^{\circ}$ (fig. 2(b)) the flow is turned due to the presence of the shock wave originating at the wedge leading edge. As the deflection angle is increased the flow is turned more and more sharply due to the increasing shock intensity. Behind the shock the flow outside the boundary layer on the bypass plate is parallel to the wedge surface, but on the surface of the bypass plate the flow is no longer parallel to the wedge surface. At $\delta_L = 10^\circ$ (fig. 2(c)) a ridge line emanating from the wedge leading edge is beginning to form. This ridge line may be due to the piling up of the oil. As δ_L is increased to 15° and 20° (figs. 2(d) and 2(e), respectively) the ridge line becomes more distinct. Behind this ridge line the flow near the leading edge at $\delta_L = 10^\circ$ is parallel to the ridge line but at 15° and 20° is actually turned toward the ridge line.

The shadowgraphs corresponding to the oil-flow photographs are presented in figure 3. Shock location is indicated by an arrow. Because of the latent fluorescence of the oil, the oil-flow streamlines are still visible. From this it is seen that, at $\delta_L = 10^\circ$ the shock and the ridge line are very close together but a larger deflection angles the ridge line is ahead of the shock.

In order to examine the flows more closely, schematic drawings are presented in figure 4 for the lower surface. Three lines are shown: (1) the line determined by the initial turning point of the streamlines; (2) the ridge line; and (3) the shock location. No deviation of the flow occurs at $\delta_L = 0$ (fig. 4(a)). At all other deflection angles the turning point is well ahead of the shock. The ridge line first becomes visible at $\delta_L = 10^\circ$ (fig. 4(c)) very near the shock location and as deflection angle is increased moves toward the line formed by the turning point.

Stanbrook[6] defines the deflection angle at which separation begins as the angle at which the oil-flow line from the root leading edge is swept at the same angle as the shock. If this definition is taken as the criteria then separation occurs between $\delta_L = 5^{\circ}$ and $\delta_L = 10^{\circ}$. This is in qualitative agreement with Stanbrook's value of 7.5° or 8°.

Since the definition of the deflection angle for separation is arbitrary, the forward movement of the ridge line might have been taken as the criteria for separation. The data of figure 4 would indicate that some phenomena occurred between $\delta_L = 10^{\circ}$ and $\delta_L = 15^{\circ}$ which caused the ridge line to move forward rather suddenly. If this forward movement is taken as the criteria for the separation angle then the deflection angle is between 10° and 15°. Czarnecki and Lord[7] in their investigation of controls on wings at supersonic speeds found the deflection angle for separation to be about 13°. Thus the deflection angle for separation determined by the forward movement of the ridge line would be in qualitative agreement with their value.

Shock wave turbulent boundary-layer interaction near deflected controls corresponds to the case where the change in flow direction through the shock is in a plane normal to the surface. The larger deflection angle of from 10° to 15° obtained in this investigation of glancing interaction between a shock wave and a turbulent boundary layer indicates that the two types of interaction are similar phenomena. The smaller deflection angle for separation would indicate the phenomena are different. More information is needed to determine the validity of either definition of deflection angle for separation.

The data obtained at a Reynolds number per foot of 4×10^6 do not indicate any effect of change in Reynolds number in the range of this investigation. Accordingly these data are not presented.

CHAPTER III

CONCLUSIONS

Deflection angle for separation was found to vary depending upon the criteria used in defining the angle. Using the criteria that the angle at which separation begins is that at which the oil-flow line from the leading edge is swept at the same angle as the shock, a deflection angle between 5° and 10° is obtained. A deflection angle between 10° and 15° is obtained when the forward movement of the ridge line is used as the criteria.

The larger deflection angle indicated that glancing interaction is similar to the shock wave turbulent boundary-layer interaction occurring on wing surfaces or near deflected controls. The smaller angle indicated the two phenomena are different.

More data at smaller increments in deflection angle in the range from 5° through 15°, along with the corresponding pressures on the bypass plate are needed to determine the validity of either definition of deflection angle for separation.

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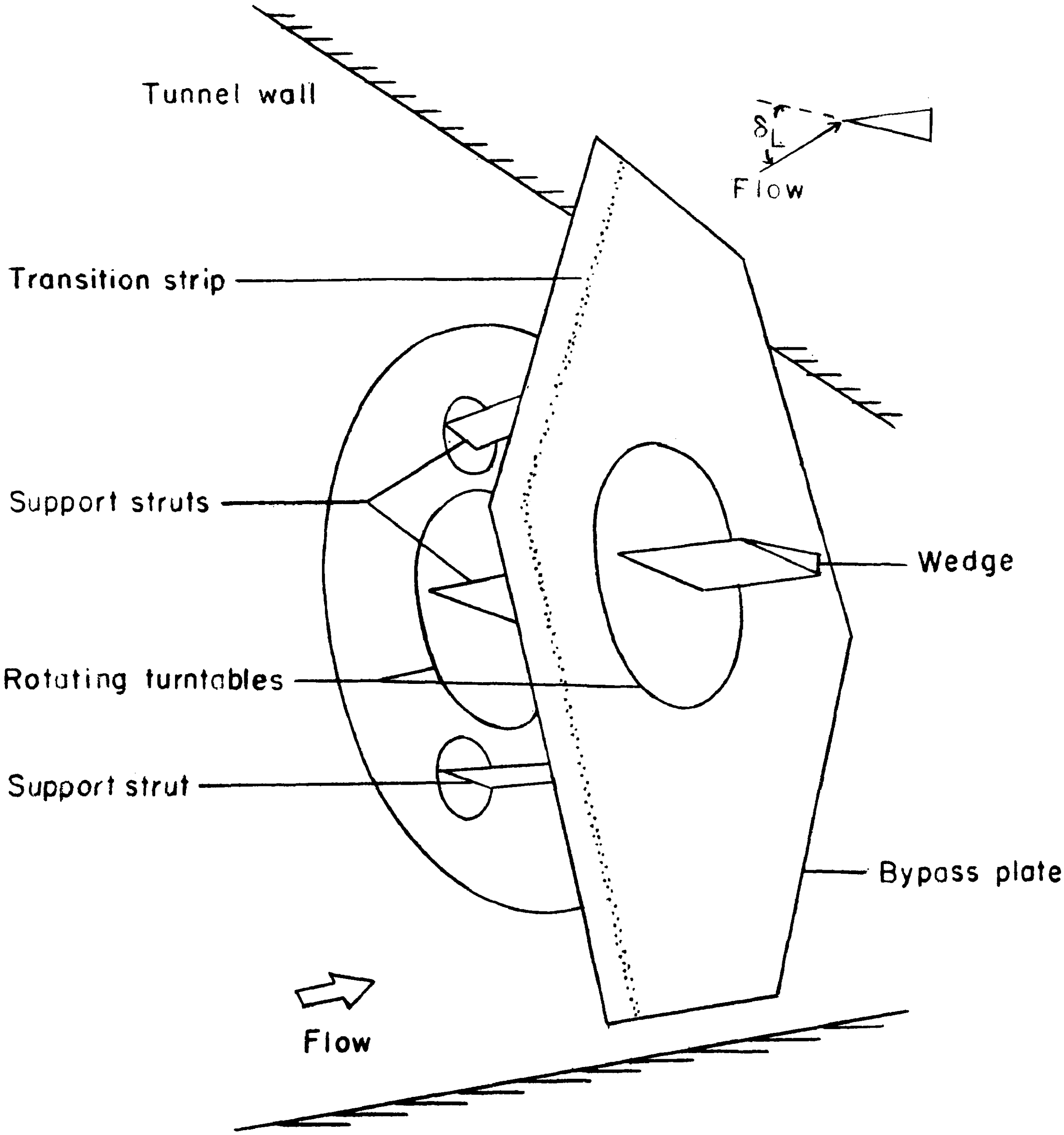
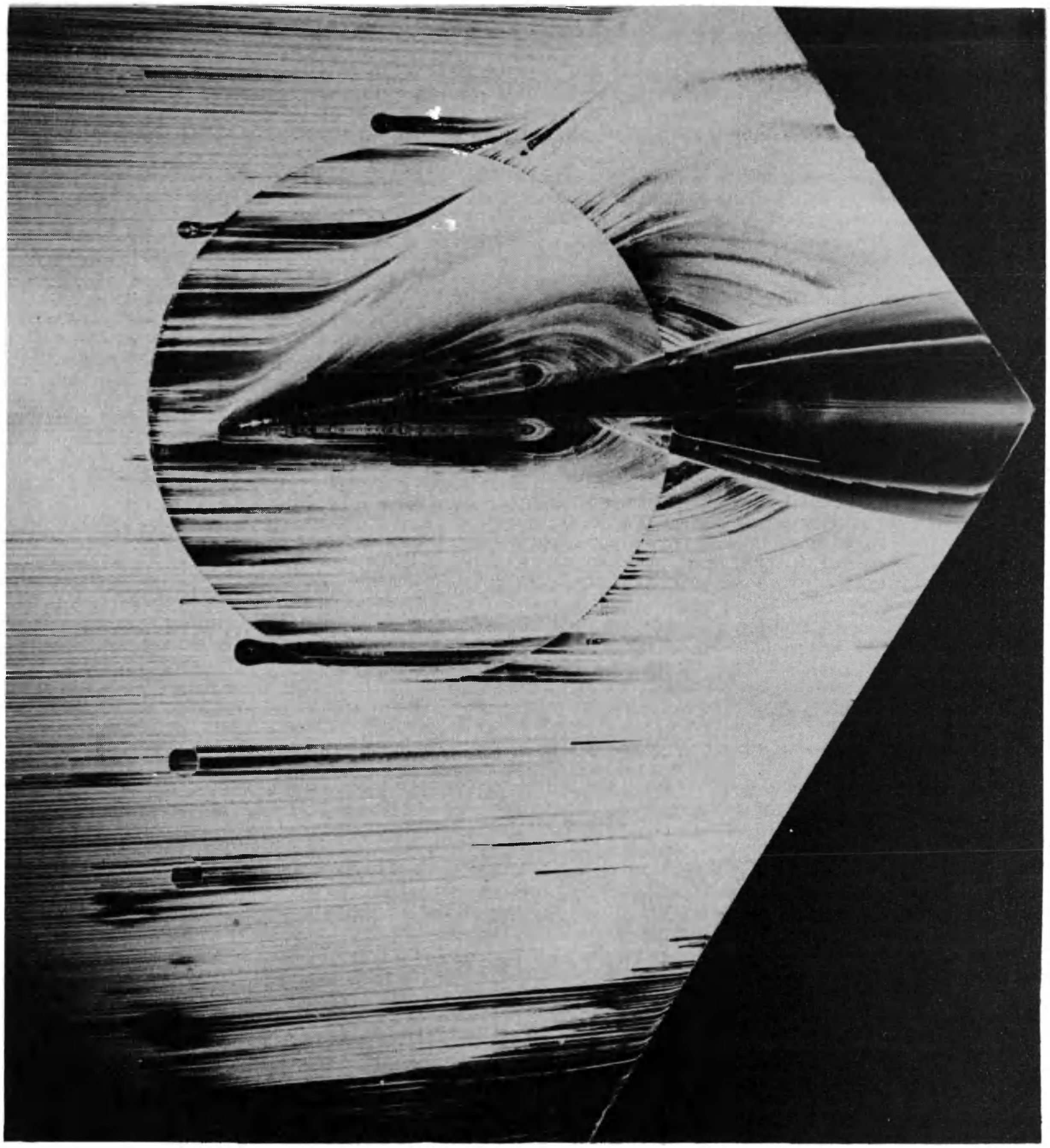
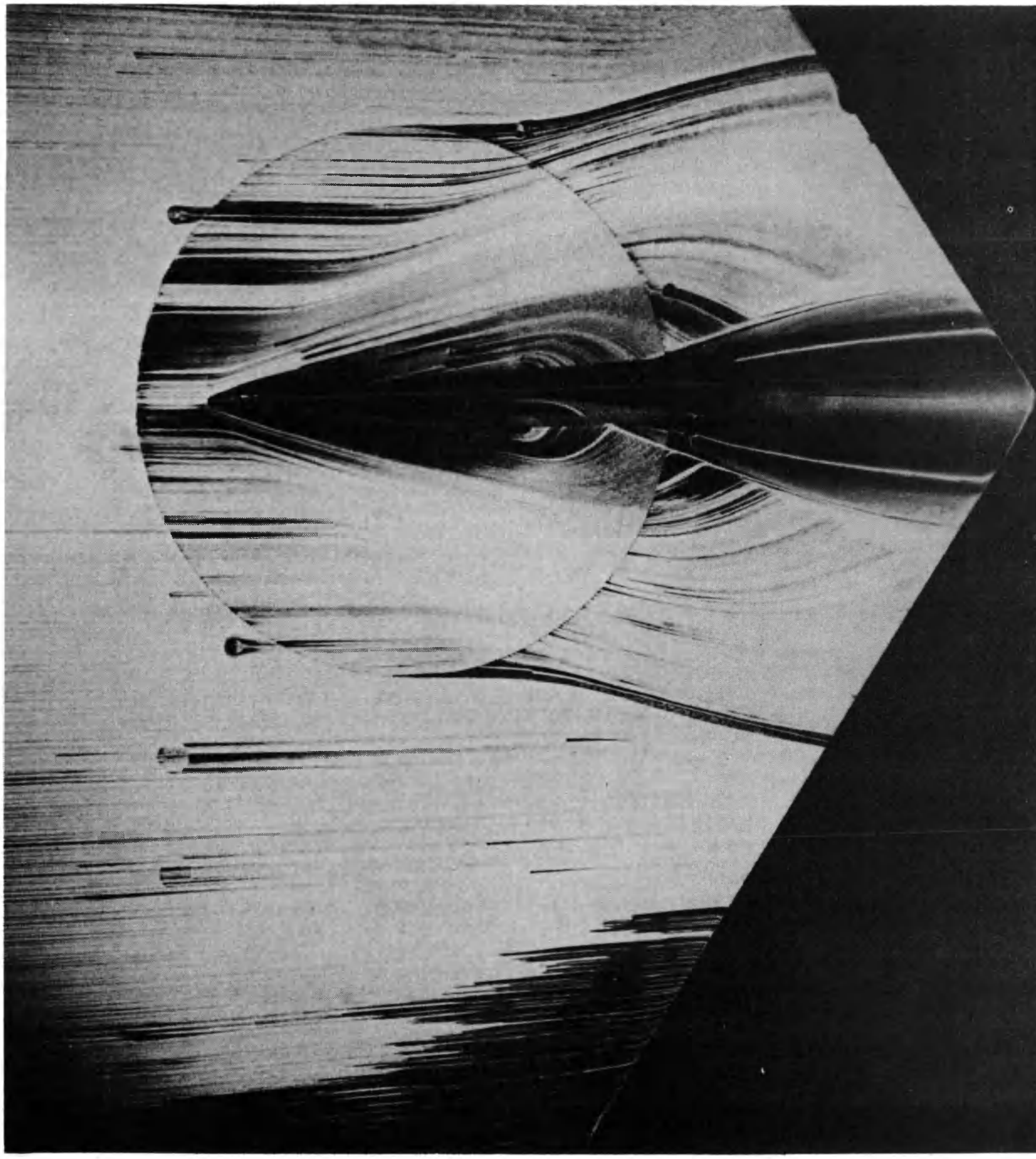


Figure 1.- Schematic drawing of test setup.



(a) $\delta_L = 0^{\circ}$. L-61-2172

Figure 2.- Oil-flow photographs.

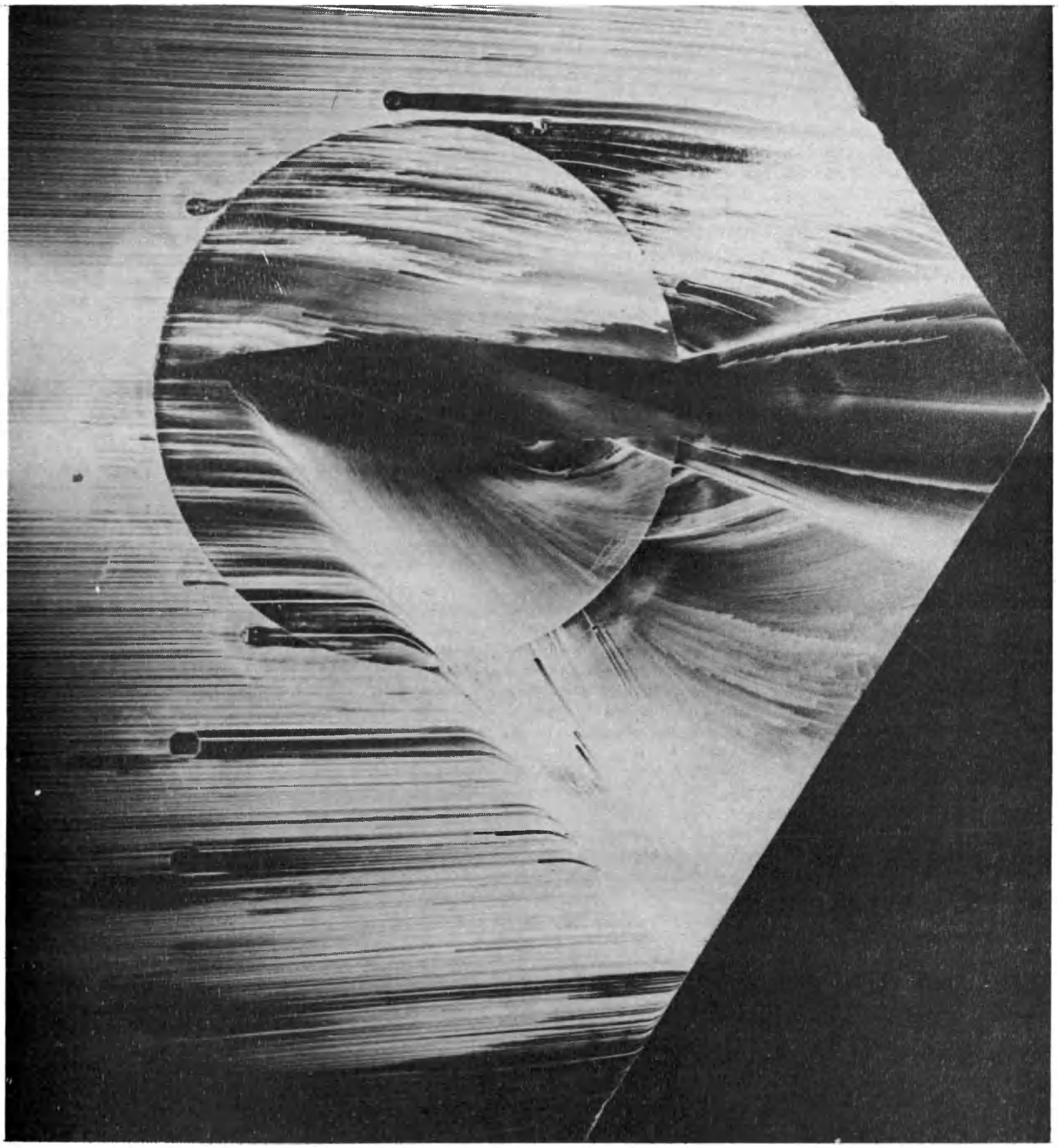


(b) $\delta_L = 5^{\circ}$. L-61-2173

Figure 2.- Continued.

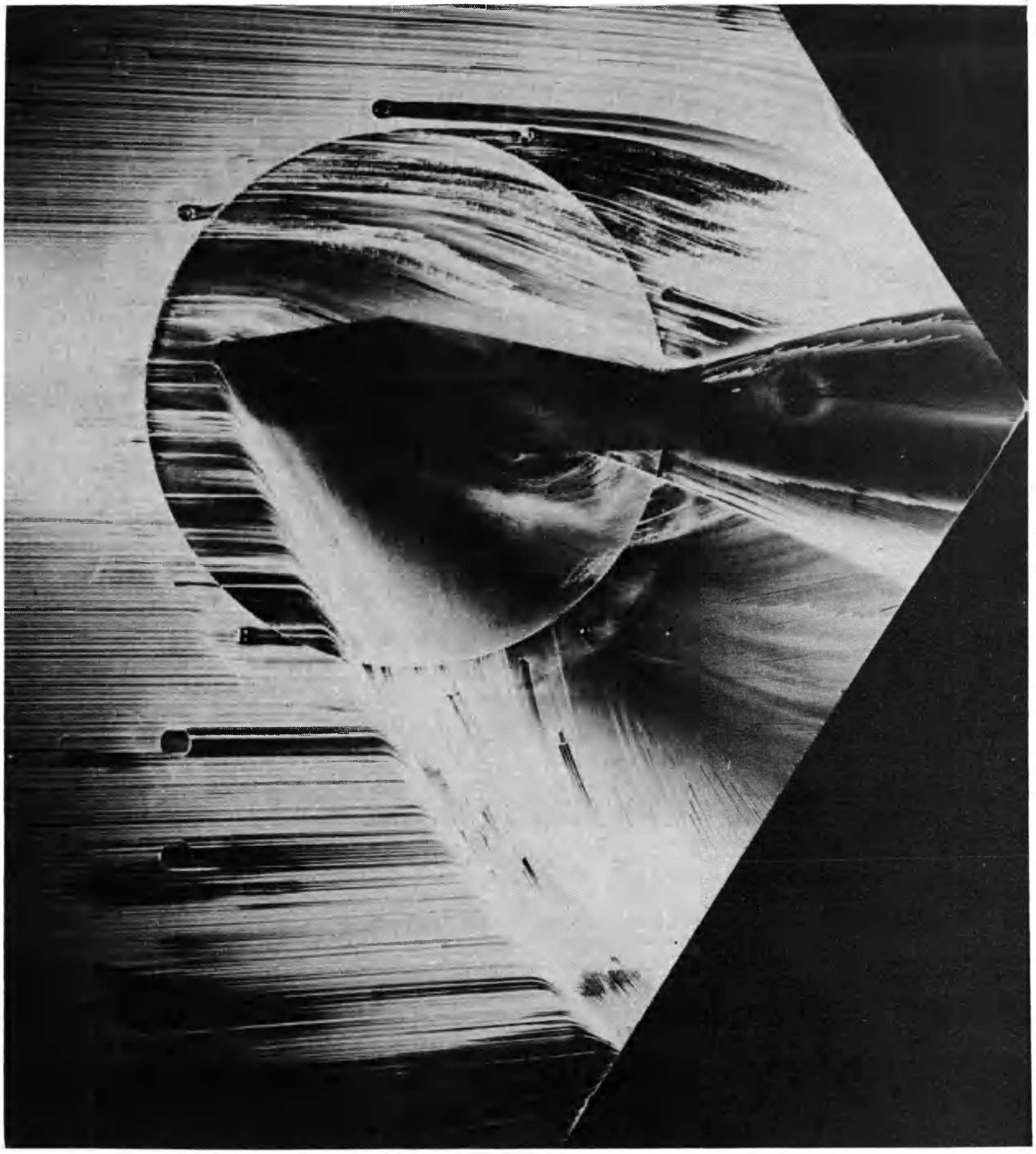
(c) $\delta_L = 10^{\circ}$. L-61-2174

Figure 2.- Continued.



(d) $\delta_L = 15^\circ$. L-61-2175

Figure 2.- Continued.

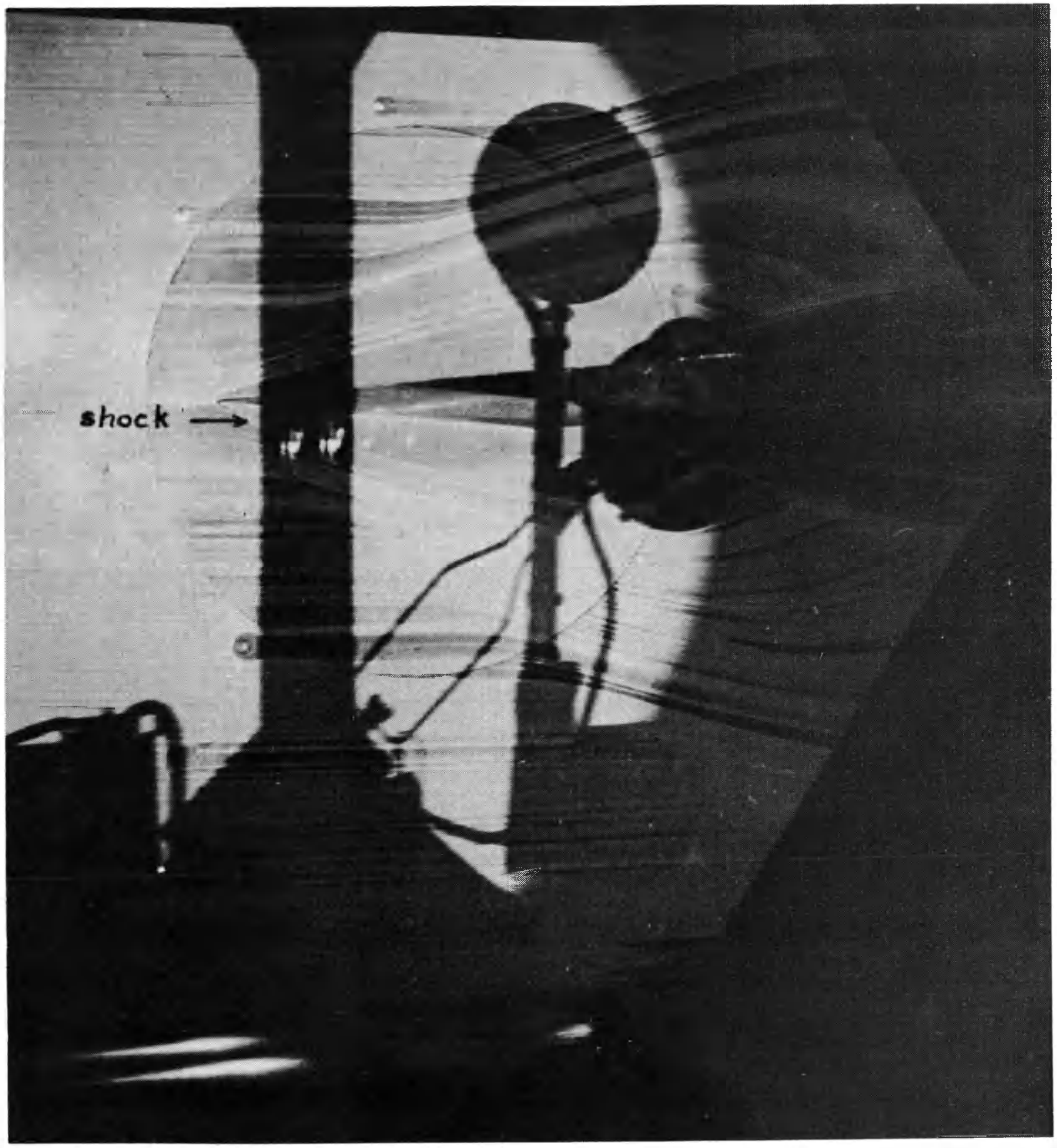


(e) $\delta_L = 20^{\circ}$. L-61-2176

Figure 2.- Concluded.

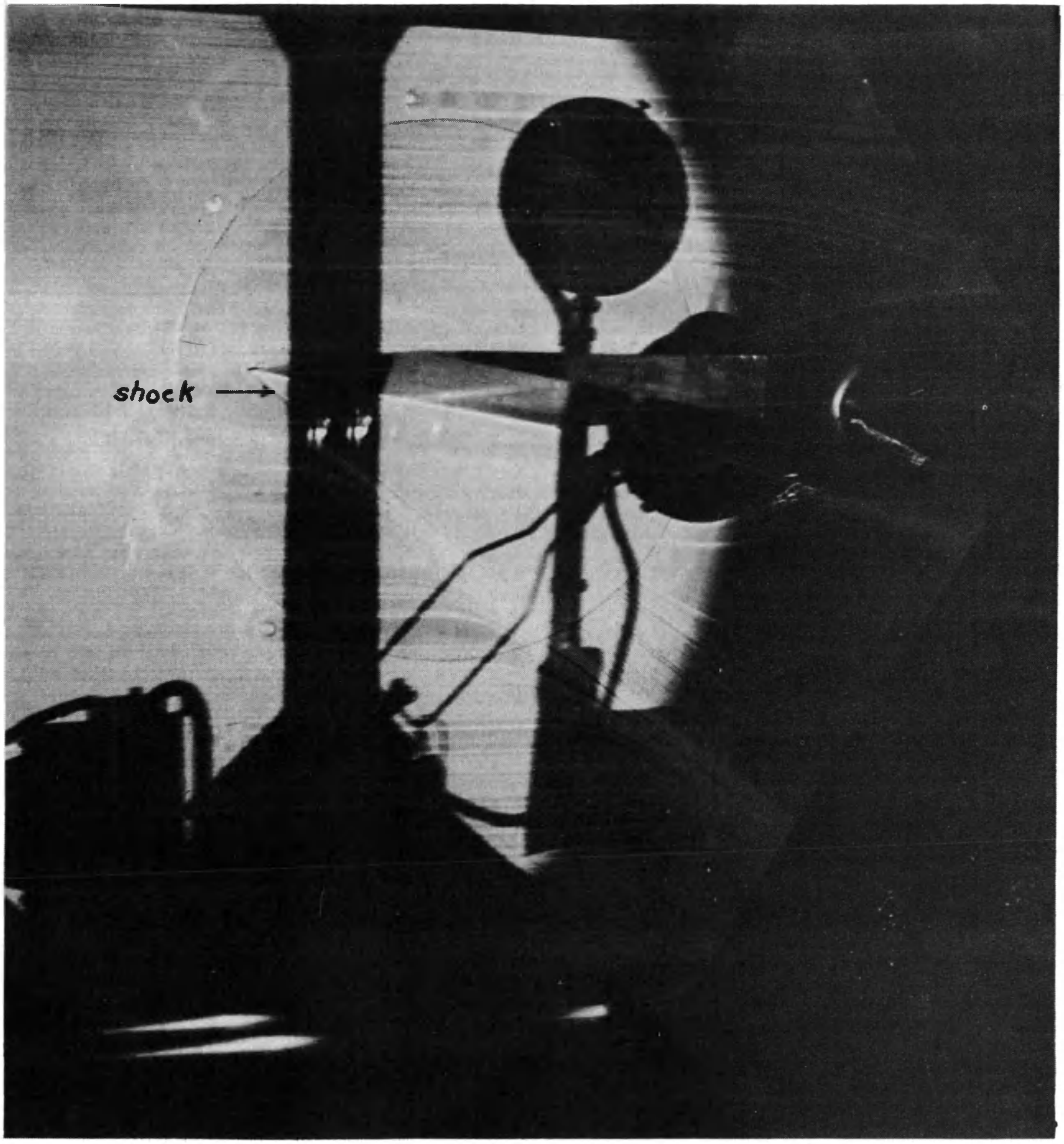
(a) $\delta_L = 0^{\circ}$. L-61-2177

Figure 3.- Shadowgraph photographs.



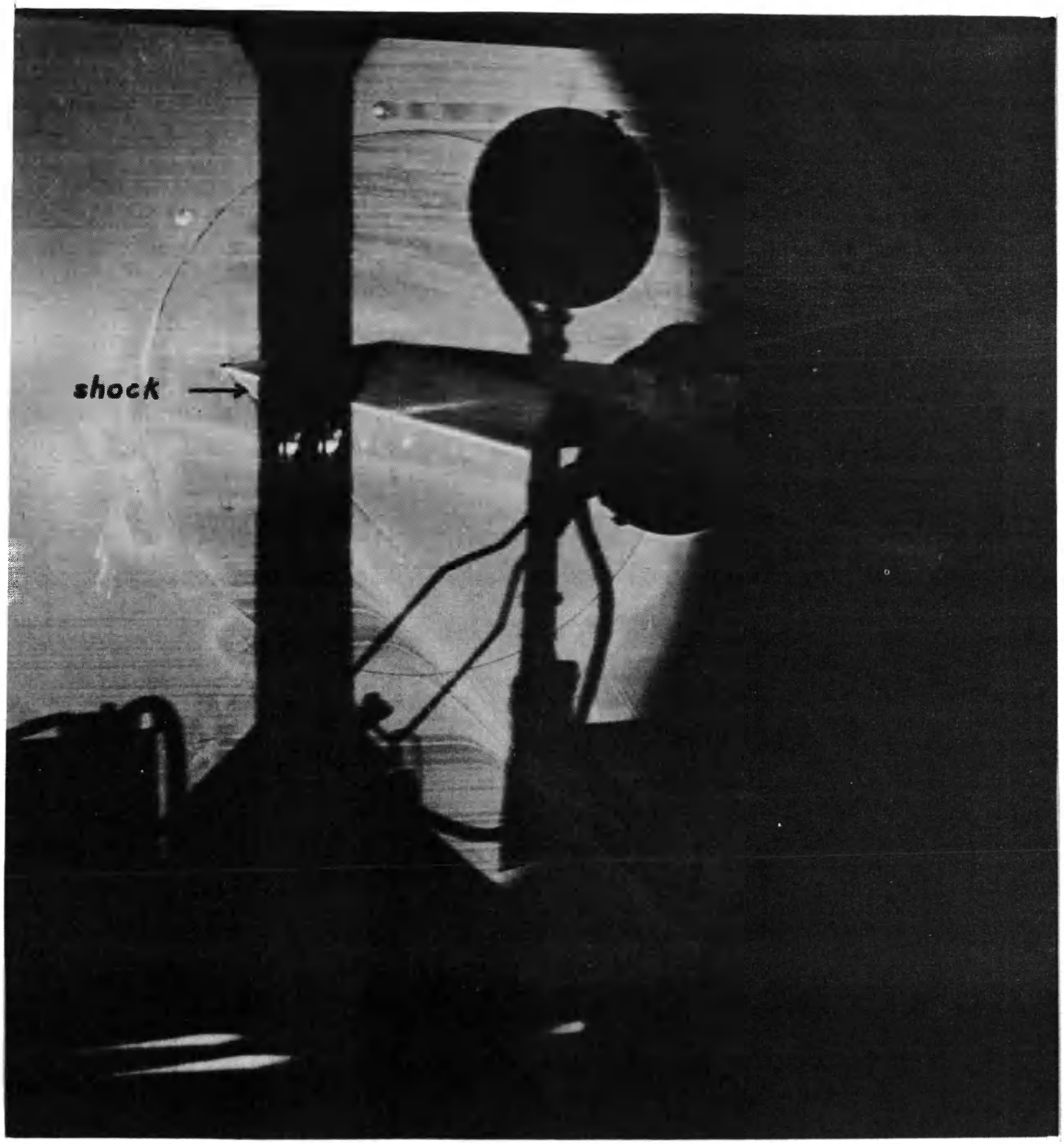
(b) $\delta_L = 5^{\circ}$. L-61-2178

Figure 3.- Continued.



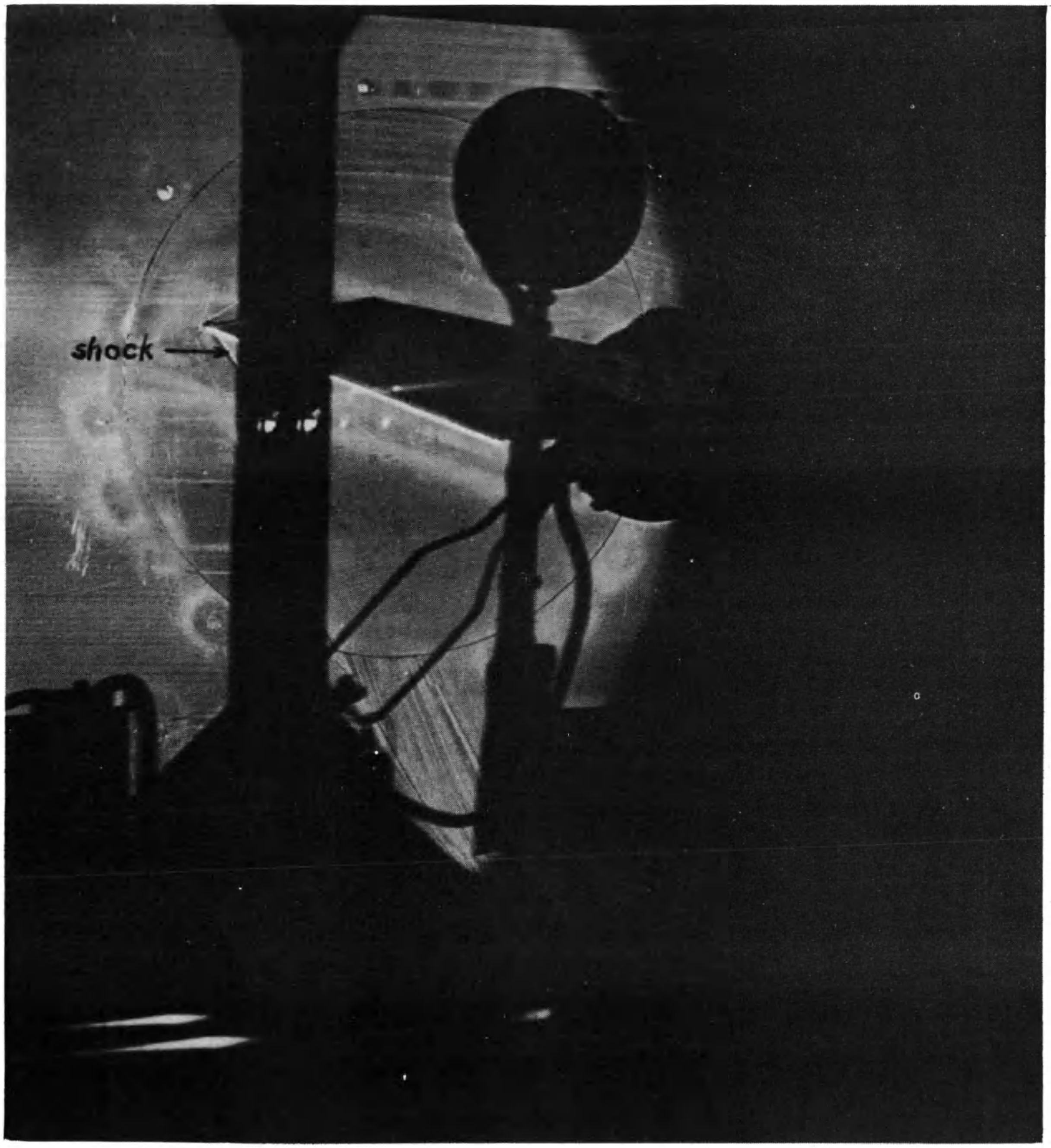
(c) $\delta_L = 10^{\circ}$ L-61-2179

Figure 3.- Continued.



(d) $\delta_L = 15^{\circ}$. L-61-2180

Figure 3.- Continued.



(e) $\delta_L = 20^\circ$. L-61-2181

Figure 3.- Concluded.

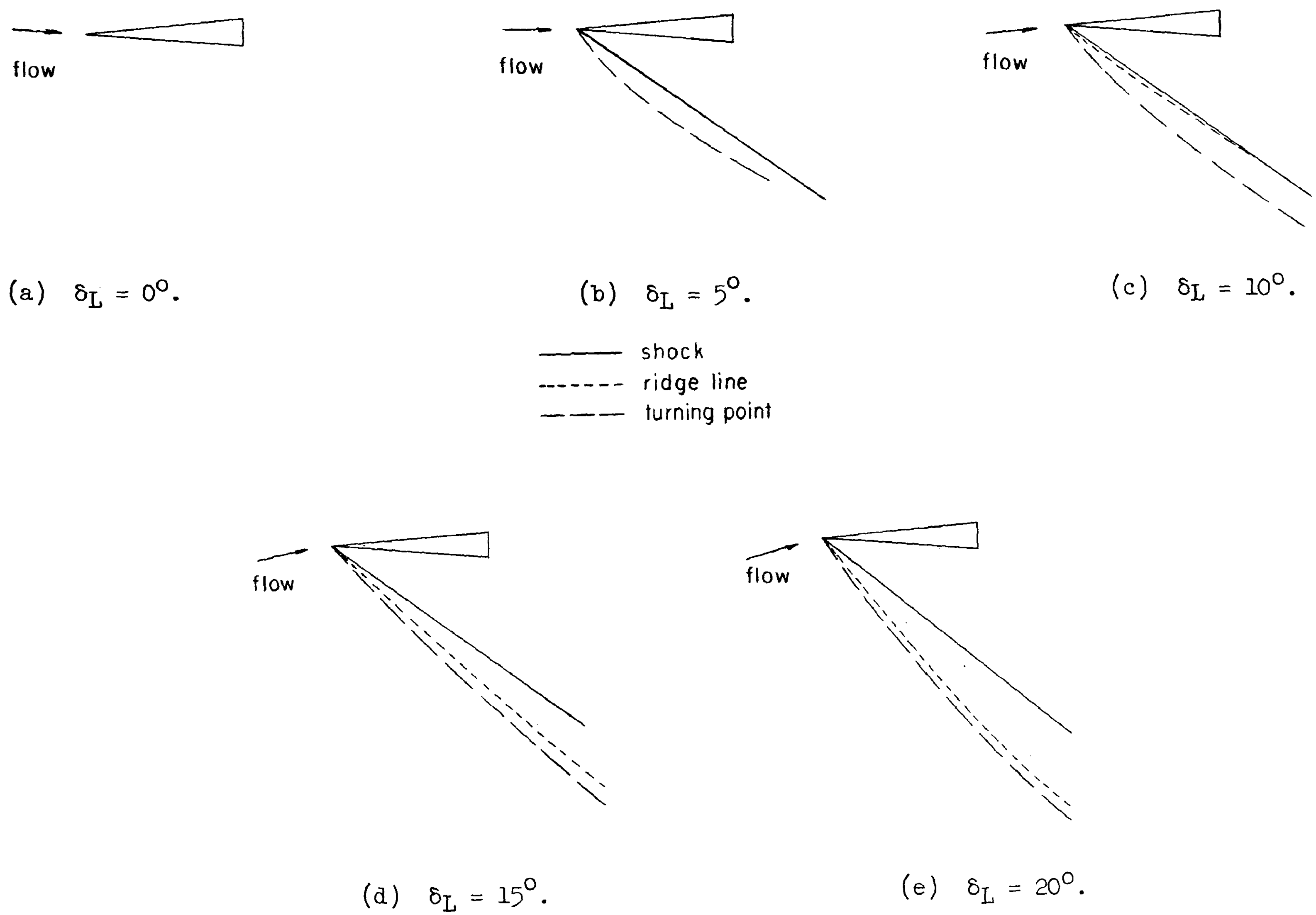


Figure 4.- Schematic drawings showing flow characteristics.

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