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An experimental study of sound diffraction at an airfoil trailing edge

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Spark photography with a sensitive schlieren system has been used to show the interaction between an incident acoustic wave and the flow around an airfoil trailing edge. Impact of the wave did not show any significant observable effect on the wake or trailing-edge boundary layer. The intensity of the wave diffracted from the edge varied considerably with the prevailing flow conditions. In the event of unsteadiness in the flow or boundary-layer separation the diffracted wave was strongly visible. In smooth flows with attached boundary layers the diffracted wave was very weak. These observations tend to support the recent conclusion of Howe (1976) that trailing-edge flows are quieter if they do not show singular behaviour. This is in contrast to the predictions of earlier theoretical models of the edge diffraction problem.

1. Introduction

The possibility that a sharp trailing edge in a turbulent flow could be a dominant noise source was shown by Ffowcs Williams & Hall (1970). In their study the effect of mean flow was not considered, and subsequent theoretical treatments of the edge diffraction problem have been primarily concerned with the effect of imposing constraints on the flow around the edge, a form of Kutta condition. On the basis of different models – convected turbulent eddy (Howe 1976), vortex-sheet instability (Crighton 1972) or a potential flow disturbance (Jones 1972) – different conclusions have been reached. Crighton (1972) predicted that suppression of the edge singularity in Ffowcs Williams & Hall's model would result in a large increase in the intensity of the diffracted field with a complete reversal in directionality. Jones (1972) found no such effect at large distances from the edge, but predicted the generation of a strong plane wave close to the diffracting plane. Davis (1975) also predicted a strong beaming effect along the wake, with a large increase in intensity. Howe (1976) has found, on the other hand, that the shedding of vorticity arising from the Kutta condition leads to a cancellation of the edge diffraction field.

This paper describes an experiment to show the interaction between the flow at the trailing edge of an airfoil and the acoustic field from a source (transducer) fixed relative to the edge. The aim was to examine three possible effects:

(i) Any effect of changes in the flow field on the directivity and intensity of the sound field diffracted from the edge.

(ii) The possibility that the incident sound field will cause flow separation at the trailing edge and a violation of the steady-flow Kutta condition, if the period of the sound field is shorter than the time required for viscous forces to act.

(iii) The possibility of exciting a boundary-layer instability as observed in jet nozzle flows (Poldervaart, Wijnands & Bronkhorst 1974). An interaction of this type is involved in the feedback-loop mechanism proposed by Tam (1974) as an explanation for the observation of discrete tones in the noise spectrum of an isolated airfoil.

The photographs shown here are taken from an extended series of single-shot photographs. With the aid of a sensitive schlieren system it was possible to visualize the incident, reflected and diffracted sound waves in addition to wake instabilities in a low Mach number flow. No fluctuation in the wake was found that could confidently be attributed to the passage of the sound wave. In the event of unsteadiness in the flow due to boundary-layer separation or other causes the edge-diffracted wave appeared strongly. If the flow was smooth and steady, with attached laminar boundary layers, the diffracted wave tended to disappear.

2. Experimental details

The experiment was conducted in a 115×165 mm intermittent wind tunnel at the Engineering Laboratory, University of Cambridge. This wind tunnel was fitted with subsonic liners and operated in the range Mach 0·15–0·55. The model was a C4 symmetrical airfoil of 100 mm chord and was mounted on a streamlined support allowing several angles of incidence between $-5{\cdot}7^\circ$ and $+11{\cdot}3^\circ$.

The optical arrangement was a Z-type schlieren system with mirrors of diameter 200 mm and focal length 2·4 m. An argon-jet spark light source of duration 0·3 μs was used and photographs recorded on 35 mm FP 4 or high-speed Ektachrome film.

The acoustic source was located in the wind-tunnel ceiling, vertically above the centre of the airfoil trailing edge and at a distance of 70 mm from it. Two types of source were used.

(i) A piezoelectric transducer 25 mm in diameter operated at 40 kHz. It was possible to generate a sound field of intensity 144 dB in short pulses. A calculation based on Raman–Nath diffraction theory (Smirnov, Kheifets & Shenderov 1973) indicated that it should be possible to visualize the sound field with the schlieren system provided that the slit light source was not more than 0·15 mm wide.

(ii) A single-pulse spark gap of energy 2 J supported at the focus of a paraboloidal reflector 95 mm in diameter. This acoustic source generates a weak spherical wave followed by an intense plane wave formed by reflexion. The second wave has an N-shaped pressure profile with a peak amplitude of 25 mbar. The spark light source could be triggered from the acoustic spark with an adjustable time lag accurate to ± 2 μs, enabling a sequence of photographs to be taken.

3. Results

Sequences of photographs were taken to show the propagation of the pressure wave from the acoustic spark source in the absence of air flow. In figure 1 (plate 1) it is seen that the initial spherical wave generates only a very weak diffracted wave (frame b) and is of no interest. The intense plane N-wave produces a cylindrical diffracted wave strong enough to be observed, in addition to the transmitted and reflected waves (figure 2). The intensity of the (directionally dependent) diffracted wave is on average some 20 dB below that of the incident wave (Candel 1973).

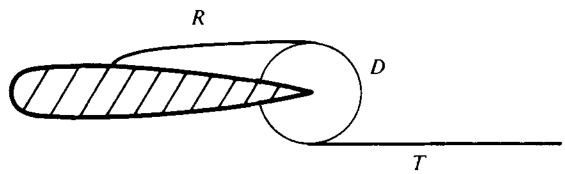

FIGURE 2. Diffraction of a plane wave at the sharp edge. T = transmitted wave; R = reflected wave; D = diffracted wave.

The sequence in figure 3 (plate 2) shows the interaction of the acoustic wave with the trailing-edge flow at Mach 0·23 (angle of incidence = $-5{\cdot}7°$). In frame (a) the plane wave is about to hit the wake. As the wave passes through, a certain amount of fluctuation is observed in the wake but it is not clear that this can be attributed to the acoustic wave; unsteadiness in the flow tended to occur when the supersonic tunnel was operated at low subsonic speeds, even without an applied sound field. What is more significant is the effect of the flow on the sound wave; the reflected N-wave has become distorted, while the transmitted wave continues relatively unaffected by the flow. In the region upstream of the trailing edge the diffracted wave has intensified compared with that in figure 1.

Intensification of the diffracted wave is even more apparent in figure 4 (plate 3), a single frame from a sequence taken at Mach 0·33. Similarly, at a more pronounced angle of incidence (11·3°) the diffracted wave also appears strongly (figure 5, plate 3; Mach 0·22).

When the airfoil was positioned at zero incidence to the flow the intensity of the diffracted wave varied considerably. At very low wind speeds the flow was generally rather unsteady, and the diffracted wave was seen clearly. But at higher wind speeds the flow was comparatively smooth and the diffracted wave became very weak (figure 6, plate 4; Mach 0·34), possibly even weaker than in the no-flow case. Figure 7 (plate 5) shows two examples from a large number of photographs taken at Mach 0·24 in order to assess the reproducibility of the flow conditions and their effect on the diffracted wave. In figure 7 (a) the diffracted wave appears strongly, whereas in figure 7 (b) it is almost invisible. Since in figure 7 (a) the wake appears to be fluctuating, whereas in figure 7 (b) it is relatively steady, there is some indication of a possible connexion between the intensity of the diffracted wave and the trailing-edge flow condition, as anticipated from the theoretical studies referred to in the introduction.

When the 40 kHz transducer was used as the acoustic source, the incident and transmitted waves could be seen clearly, but it was difficult to observe the comparatively weak diffraction field. In figure 8 (plate 6; 5·7° incidence, Mach 0·17) it is just visible. In the absence of air flow the diffraction field was totally invisible.

It was also possible to observe a diffraction field at the leading edge (figure 9, plate 6; Mach 0·34). In this experiment the transducer was mounted directly above the centre of the leading edge, the incident acoustic field radiating vertically downwards as before. As in figure 8 the diffraction field appears only in the presence of both an incident sound field and air flow. A number of photographs taken at higher wind speeds and at 11·3° incidence without sound showed strong pressure waves radiating from the leading edge, but these probably resulted from stall-induced vibration of the airfoil.

4. Discussion

The photographic study has shown that unsteady trailing-edge flow can be an important noise source. Sequences showing the propagation of the wave from the acoustic spark in still air indicated that the diffracted wave was quite reproducible, whereas in the presence of air flow the intensity of the diffracted wave varied considerably. In particular it was found that (i) with the airfoil positioned at 5·7° incidence or more the diffracted wave was intensified, (ii) at zero incidence and moderate wind speeds (Mach 0·35 or higher) the diffracted wave tended to disappear and (iii) at zero incidence and low wind speeds, the intensity of the diffracted wave appeared to increase or decrease according to whether the wake was unsteady or steady. The random fluctuations in the wake were probably associated with disturbances caused by the airfoil support.

Results (ii) and (iii) indicate that the diffracted wave intensity may depend on the steadiness of the flow in the vicinity of the trailing edge. On the other hand, result (i) suggests that the intensification of the diffracted wave may be connected with separation of the boundary layer. Close examination of the trailing-edge flow in figure 10 (plate 7; unexcited flows at Mach 0·45) indicates that for the symmetrical C4 airfoil the suction side boundary layer remains attached almost as far as the trailing edge at 3° incidence or less (figure 10*a*), whereas at 5° incidence or more (figure 10*b*) separation has occurred.

As described in the introduction, large changes in diffraction field intensity have been predicted according to the degree of edge singularity admitted in the potential flow model, the singularity being removed by imposing a form of Kutta condition. It is of interest to ascertain whether the variations in diffracted wave intensity observed in the present experiment are connected with variations in the extent to which the trailing-edge flow shows singular behaviour. This can be deduced qualitatively from observations of the wake curvature, provided that the Kutta condition can be interpreted in the context of an acoustically perturbed flow with viscous boundary layers present.

The experimental conditions are essentially high frequency in the sense that the airfoil chord is much larger than the effective wavelength of the sound. From an acoustical point of view it is then appropriate to consider the theory of scattering by a semi-infinite screen, for which the Kutta condition was defined by Orszag & Crow (1970). In describing unsteady flows on one or both sides of a semi-infinite thin plate they specified (i) a *full* Kutta condition requiring the trailing vortex sheet always to leave tangentially to the plate, and (ii) a *rectified* Kutta condition requiring the vortex sheet to leave the plate in such a way that no fluid turns through an angle greater than π. This is the Kutta condition referred to by Crighton and others; it is essentially an artificial device to introduce viscosity into an inviscid solution, whereas the more usual airfoil problem is concerned with the determination of circulation.

For a real flow around an airfoil, with boundary layers present, the circulation is not determined by the Kutta condition, but by viscous effects at the trailing edge. Nonetheless it is found that to establish maximum lift the dividing streamline must leave the trailing edge tangentially (Thwaites 1960). In the case of a wedge-shaped trailing edge, the condition proposed by Thwaites is that the tangent to the dividing streamline should pass through the interior of the airfoil. As a constraint on the direction of the flow, this trailing-edge condition is equivalent to the rectified Kutta condition of Orszag & Crow; however, it is strictly applicable only to unseparated flow.

When separated boundary layers are present the only trailing-edge condition that can be invoked to determine the lift is the requirement that the total convection of vorticity from the boundary layers into the wake should be zero (Sears 1976). For an airfoil at moderate incidence, with one boundary layer separated (figure 10*b*), it is most unlikely that this trailing-edge condition is valid. In steady flow the total vorticity flux due to convection and diffusion (out of a closed surface surrounding the airfoil) is zero. Since a separated boundary layer diffuses some of its vorticity into the free stream while an attached one does not, the fluxes of vorticity convected into the wake from the two boundary layers will not be equal and opposite. Consequently, when a boundary-layer separation is observed (figure 10*b*) the trailing-edge condition is not the ideal condition that determines maximum circulation and lift. The same applies to an unsteady flapping wake (figure 7*a*). Qualitatively, these trailing-edge flows can be said to show a greater degree of singularity than the smooth flows (figures 7*b*, 10*a*) in the sense that the fluid in the pressure side boundary layer is attempting to negotiate a sharp turn around the trailing edge.

In conclusion, it is possible that the observed intensification of the diffracted wave in figures 3–5 is related to the non-ideal trailing-edge condition associated with separation of the boundary layer. In figures 6 and 7(*b*) there is steady flow with attached boundary layers, and the weakening of the diffracted wave may be connected with the smooth flow at the trailing edge. These results give some support to the conclusions of Howe (1976), although the experimental conditions correspond more closely to the model assumed by Jones (1972).

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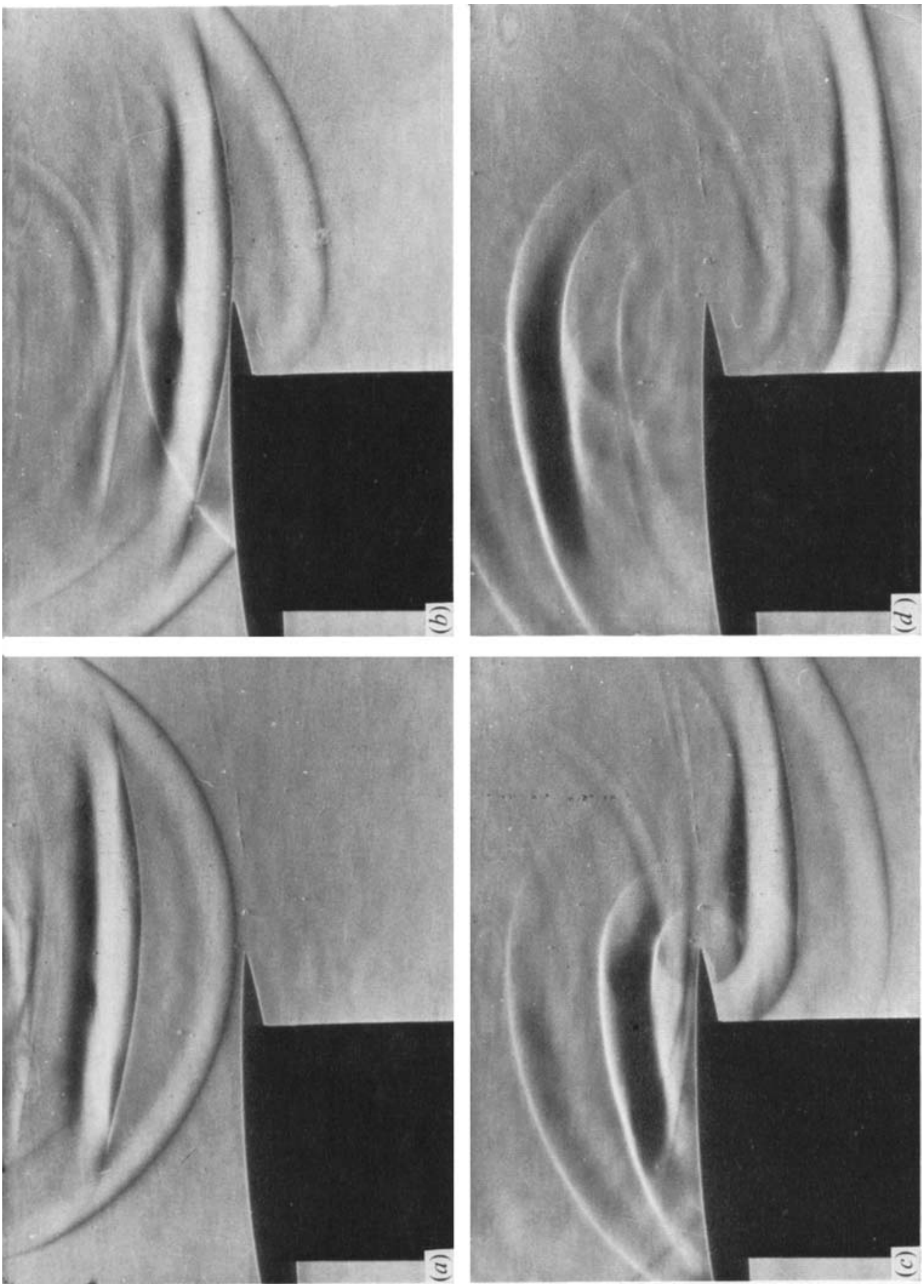

FIGURE 1. Propagation of the sound wave from a spark source situated in the wind-tunnel ceiling, and its reflexion and diffraction at an airfoil trailing edge, in the absence of air flow. The initial spherical wave is weak; it is followed by an intense plane wave with an N-shaped pressure profile. Time between successive frames = 50 μs; schlieren knife edge horizontal, opaque side uppermost. In this and all subsequent figures the air flow is from left to right, and the incident sound waves are propagating vertically downwards.

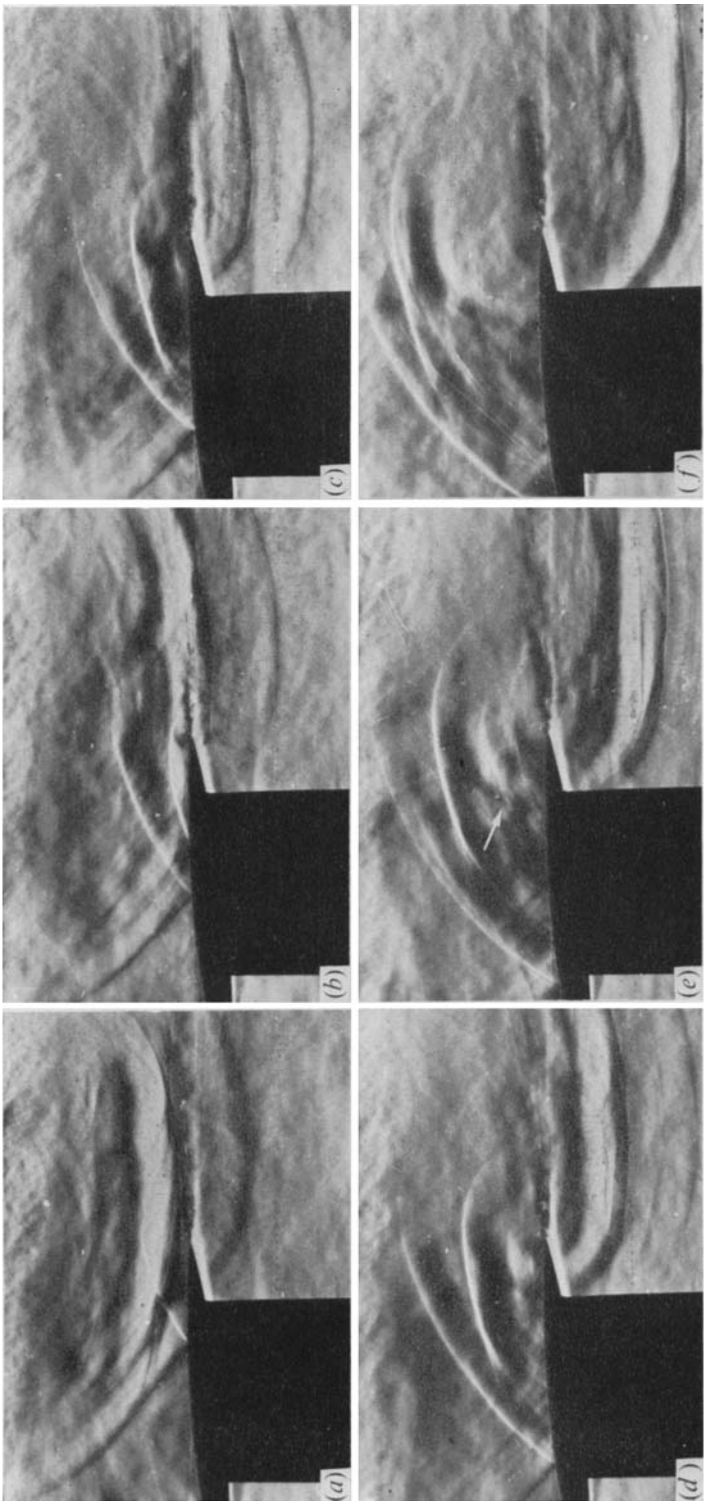

FIGURE 3. Diffraction of the plane acoustic wave in the presence of air flow at Mach 0·23. In frame (*a*) the wave is about to hit the wake; frames (*d*)–(*f*) show the propagation of the relatively strong diffracted wave (arrowed in frame *e*). Angle of incidence = $-5{\cdot}7°$; time between frames = 20 μs; knife edge horizontal, opaque side uppermost.

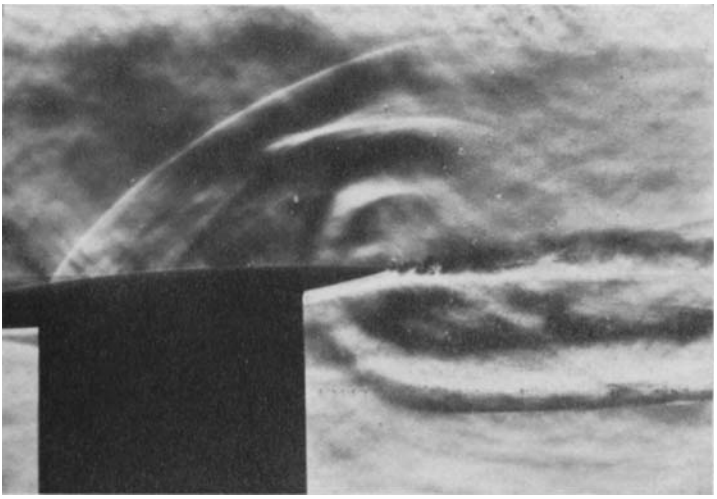

FIGURE 4. Further intensification of the diffracted wave at Mach 0·33. Angle of incidence = $-5{\cdot}7°$; knife edge horizontal, opaque side uppermost.

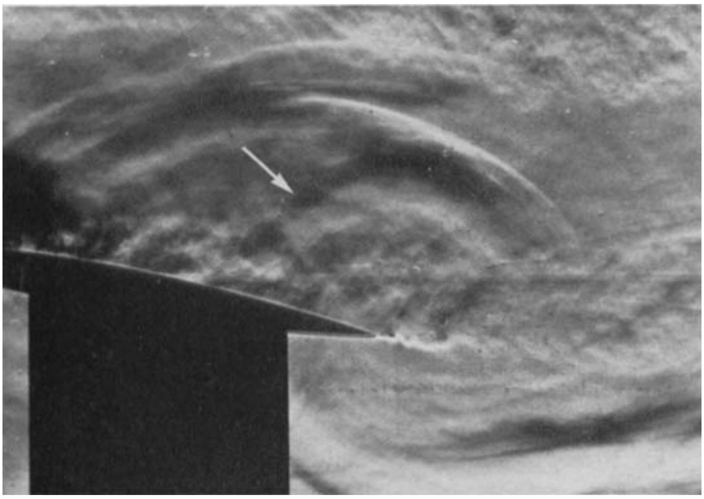

FIGURE 5. Appearance of the strong diffracted wave (arrowed) as it emerges from the turbulent boundary layer at 11·3° incidence, Mach 0·22. Knife edge horizontal, transparent side uppermost.

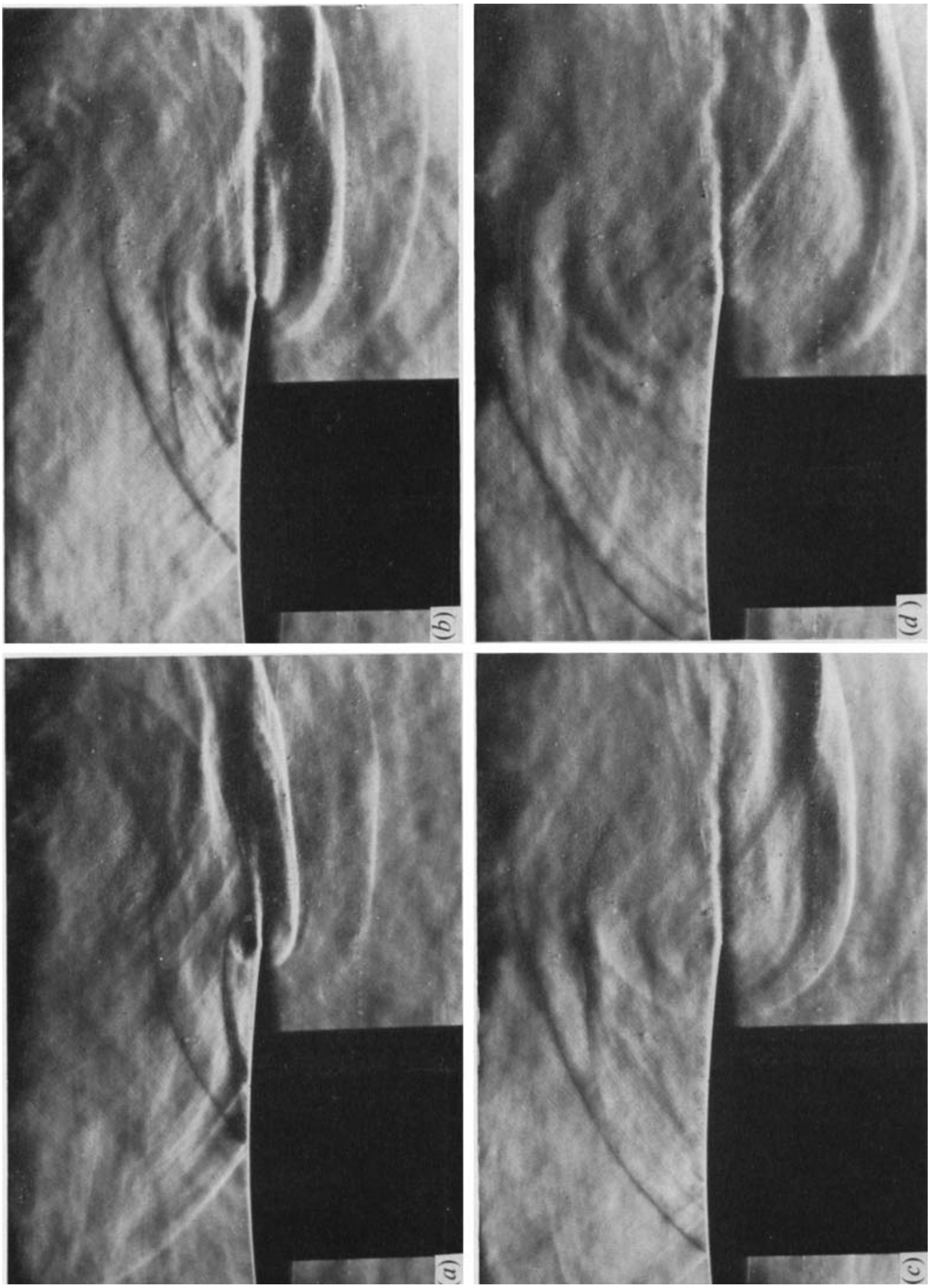

FIGURE 6. Diffraction of the plane wave at zero incidence, Mach 0·34. In frame (*a*) the incident wave is passing through the wake. The diffracted wave in frames (*b*)–(*d*) is comparatively weak. Time between frames = 25 μs; knife edge horizontal, transparent side uppermost.

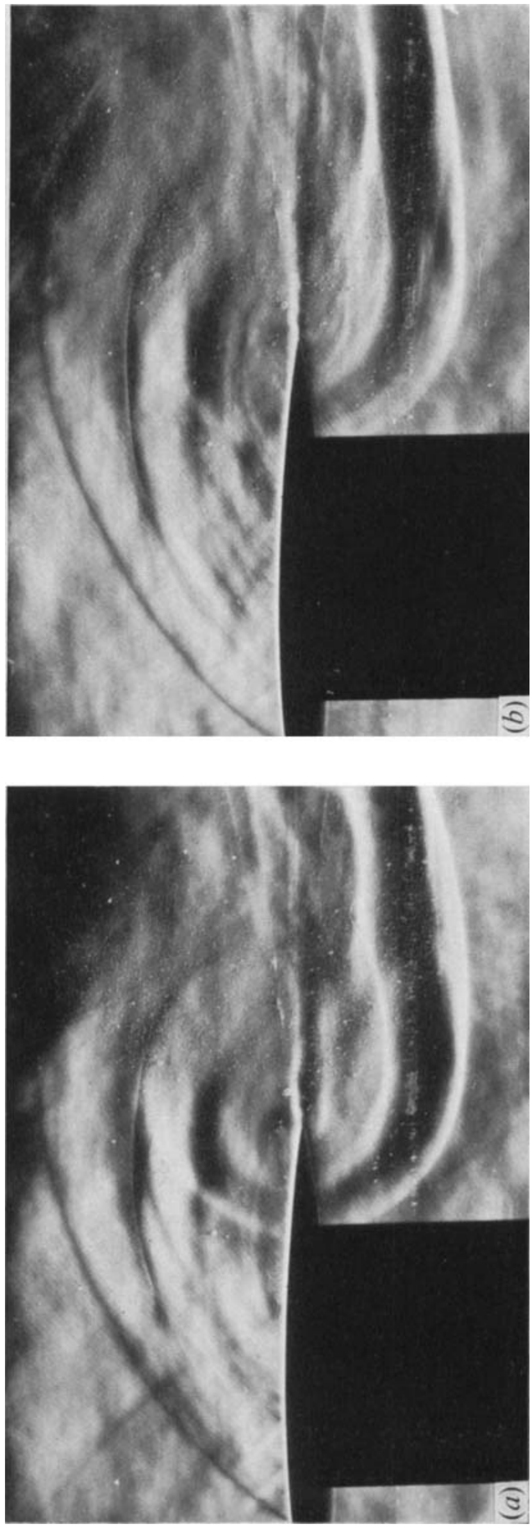

FIGURE 7. Two photographs triggered at the same delay time at zero incidence, Mach 0·24. In (*a*) the diffracted wave is strong. In (*b*) the flow is more steady and the diffracted wave is weak. Knife edge horizontal, transparent side uppermost.

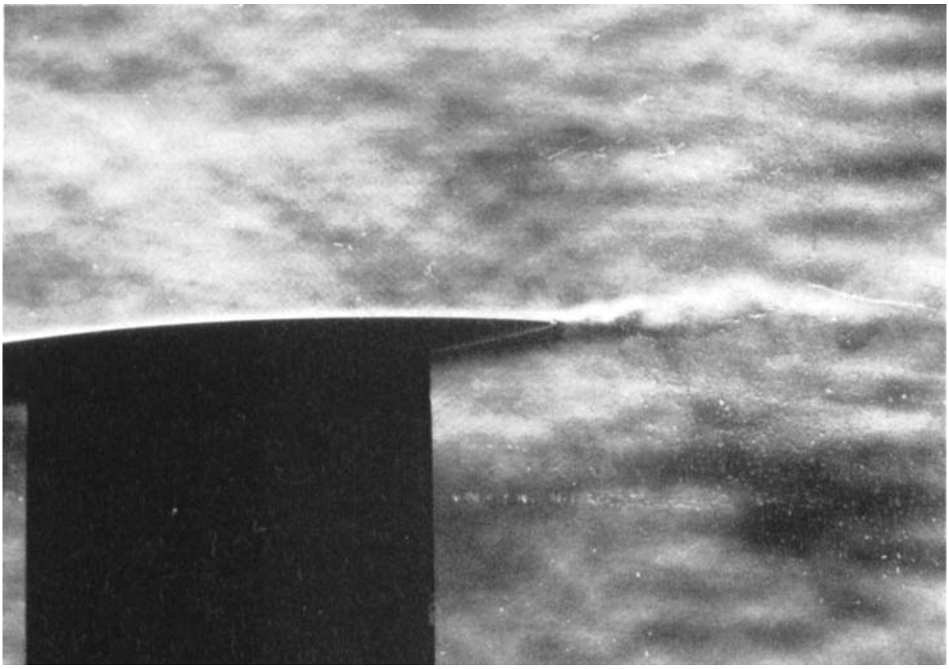

FIGURE 8. Diffraction of the 40 kHz sound field from a transducer mounted in the wind-tunnel ceiling at $-5{\cdot}7°$ incidence, Mach 0·15. The diffraction field is strongest in the direction 45° upstream of the trailing edge. The waves to the right are reflexions from the tunnel floor. Knife edge horizontal, transparent side uppermost.

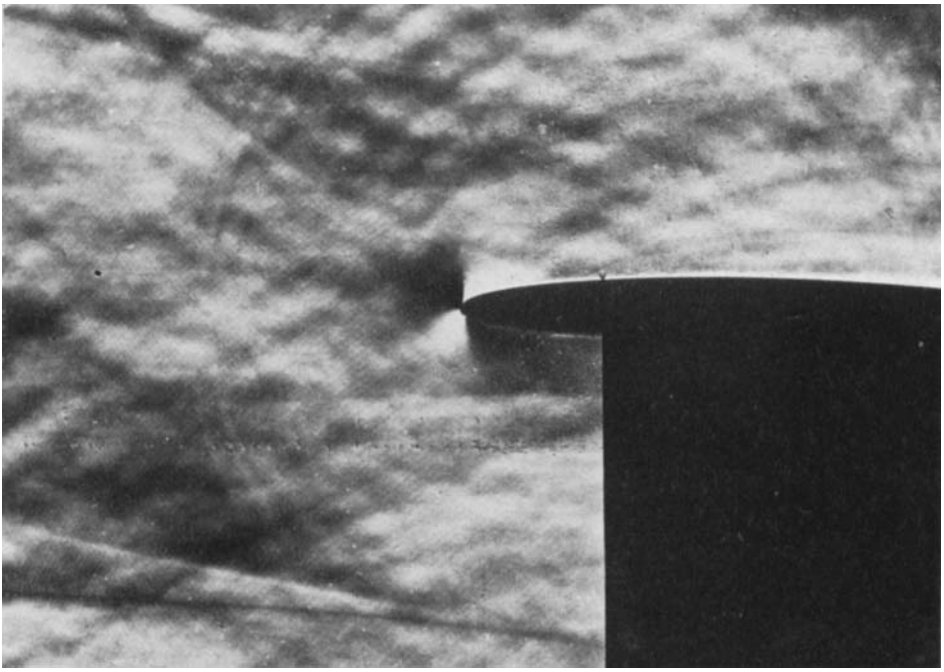

FIGURE 9. Diffraction of the 40 kHz sound field from a transducer mounted in the tunnel ceiling above the leading edge. Angle of incidence zero; flow speed Mach 0·34; knife edge horizontal, transparent side uppermost.

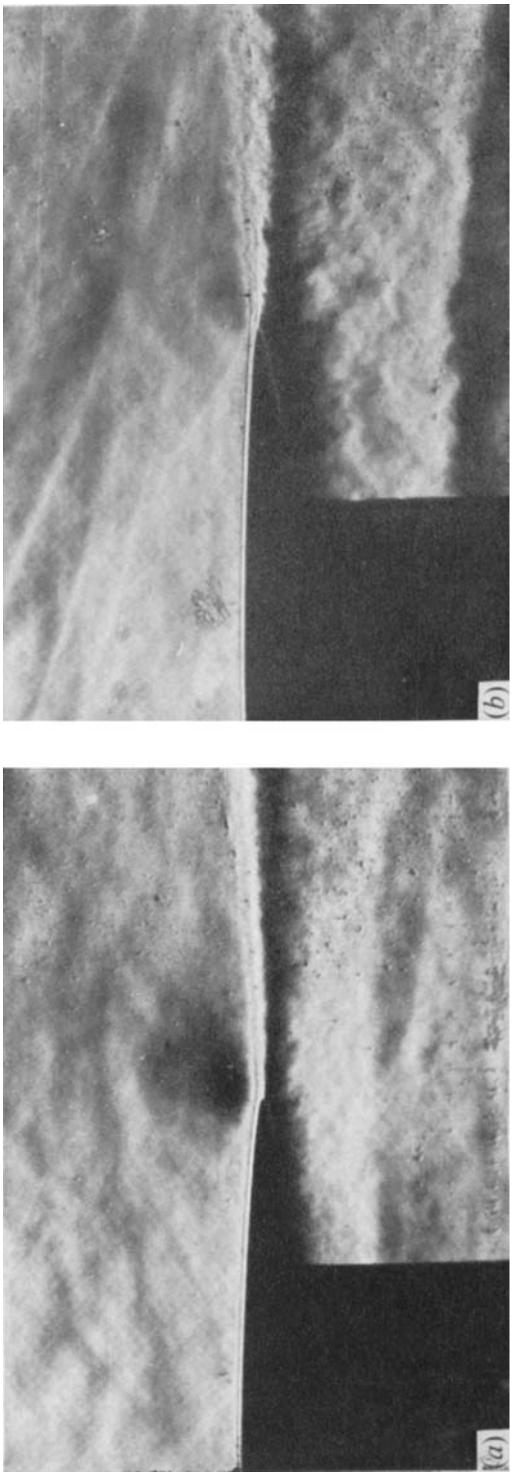

FIGURE 10. Close-up of the boundary layer and near wake at Mach 0·45 (no sound field applied). In (*a*) the angle of incidence is $-2{\cdot}9°$ and the boundary layers remain substantially attached. In (*b*) the angle of incidence is $-5{\cdot}7°$ and separation has occurred. Knife edge horizontal, transparent side uppermost.