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Note on the Use of the Three-Dimensional Shock-wave Recorder for Studying Interference in a Supersonic Wind Tunnel

By
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Note on the Use of the Three-Dimensional Shock-wave Recorder for Studying Interference in a Supersonic Wind Tunnel

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Summary.—A description is given of the three-dimensional shock-wave recorder which has been used to investigate wind-tunnel interference on a sting-mounted wing having 50 deg of sweepback.

Diagrams of the shock-wave patterns round the wing are given for the following conditions of Mach number and incidence :—

$M = 1 \cdot 4$ Incidence = 4 deg
$M = 1 \cdot 6$ Incidence = 4 deg
$M = 1 \cdot 8$ Incidence = 0 deg

From these diagrams it can be inferred that the outboard portion of the particular wing tested suffers interference from the tunnel walls at $M = 1 \cdot 4$. At the higher Mach numbers the model behaves effectively as if it were in free air.

In order to use the three-dimensional shock-wave recorder, the wind-tunnel working-section requires a window whose length is at least three times the tunnel width and which extends equal distances ahead of and behind the model. This is a very serious drawback to the general use of the method. The analysis of the photographic records is straightforward, though rather tedious.

1. *Introduction.*—In supersonic wind-tunnel testing it is generally necessary to use a model small enough to avoid wind-tunnel interference within the range of incidence to be covered. When interference is present, it may have large effects on the characteristics of the model, for which it is usually impossible to correct.

Interference from the tunnel walls may take two forms. Firstly, disturbances from the front of the model, propagated to the walls, may be reflected on to the model further back. Secondly, tunnel choking may occur when the constriction in the tunnel due to the model is too large. Both forms of interference generally cause an increase of pressure over part of the model, and their front boundaries are shock-waves. If there is no choking and there are no reflected shock-waves falling on to the model, then the flow over the model is effectively the same as that occurring in an unbounded air stream at the same Mach number and Reynolds number.

For two-dimensional wing tests, a single schlieren or shadowgraph photograph, taken by using a beam of light passing through the tunnel parallel to the wing span, will provide the information necessary for judging whether interference is present. For tests on three-dimensional models,

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such as swept-back wings, a three-dimensional picture of the shock-waves is necessary. In some circumstances it is possible to make the shock-waves visible by placing a light in the settling chamber and running the tunnel with undried air[2]. To make a plotting of the shock-waves produced with dry air, a three-dimensional shock-wave recorder has been developed by Dr. F. E. Lamplough of I.A.P. Department, Royal Aircraft Establishment[1]. This instrument has been used to obtain the shock-wave patterns produced by a swept-back wing in a tunnel of rectangular section.

The shock-wave recorder has a further use for fundamental research into supersonic flow around wings. So far, most theoretical investigations into the characteristics of wings have been based on linear theory which assumes that all disturbances are small, and that they affect the airflow only along and behind Mach lines from the origin of the disturbance. Real wings of finite thickness and at finite incidence produce disturbances which may be propagated through the fluid ahead of the Mach lines. Finite disturbances which cause a compression of the fluid produce shock-waves, which can be recorded. A study of the positions of shock-waves about a wing, which is free from tunnel interference, may help towards indicating errors in theoretical work due to the assumptions of linear theory.

2. *Test Equipment.*—2.1. *Three-dimensional Shock-wave Recorder.*—The three-dimensional shock-wave recorder, developed by Dr. F. E. Lamplough is described briefly. It is illustrated diagrammatically in Fig. 2.

Light from a point source, passing through a lens system gives a parallel beam which travels through the working-section of the tunnel. At the other side of the tunnel the light falls on to a screen containing a movable slit. Light which has crossed the working-section in a thin band parallel to the plane of the wing passes through the slit, behind which is a photographic film. Moving the slit perpendicular to its length alters the distance above or below the model of the band of light which is allowed to fall on to the film.

This system is arranged to rotate about an axis perpendicular to the plane of the wing, so that the beam passing through the tunnel can move through about 50 deg on either side of the perpendicular to the tunnel walls. As the apparatus rotates, the photographic film is moved behind the slit in such a way that there is a fixed relation between the angular movement of the beam and the linear movement of the film. A line down the film produced by a wire across the slit indicates the centre of the beam. The apparatus rotates about an axis passing through the centre of the beam.

One frame of film just gives information about the shock-wave positions in a plane parallel to that of the wing. A typical frame is shown in Fig. 4. Two vertical wires are fixed at either side of the tunnel so that a line joining them is perpendicular to the tunnel walls. These wires produce two shadows on the film. The point of their crossing indicates the strip of film which was under the slit when the beam was perpendicular to the air stream. By knowing the ratio of angular movement to linear movement of the film, the angular position of the beam represented by any strip across the film can be found.

A shadow is produced in a parallel beam, at points where light is tangential to a shock-wave. This is illustrated in Fig. 3. Thus an indication of a shock-wave on the film at Point A (Fig. 4) shows that when the beam makes an angle α to the tunnel walls, a line b units of length from the centre of the beam is tangential to a shock-wave. Thus the film gives the information necessary for plotting the tangents to the shock-waves. Having plotted a series of tangents, their envelope giving the actual shape of the shock-wave can be sketched.

The whole apparatus can be tilted about a horizontal axis so that the plane in which the shock-waves are photographed may be either parallel to the direction of the airflow, or, if the wing is at incidence, parallel to its plane.

The process of converting films to obtain the actual position of the shock-waves is simplified by using a special ' reduction box '. This device enables the tangents to the shock-waves to be plotted directly from the film, without intermediate computation. A correction for the displacement of the light as it passes through the glass walls of the tunnel is applied automatically by the ' reduction box '.

2.2. *Model and Wind-Tunnel Data.*—Fig. 1 shows the general arrangement of the wing in the tunnel, for which results are presented in this note.

The wing, made of brass, was untapered, and had a sweepback of 50 deg ; its sections were symmetrical, 10 per cent thickness/chord ratio, and consisted of two circular-arcs. The span and chord were 2·2 in. and 0·66 in. respectively. The wing was mounted on a sting, and located 1·2 in. from the floor of a tunnel of 3 in. square cross-section. The vertical tunnel walls were of specially selected, optically flat, glass.

3. *Details of the Tests.*—Shock-wave photographs were taken at Mach numbers of 1·4, 1·6 and 1·8 for a range of incidences.

All the photographs were of the flow either in the plane of the wing or beneath it. The flow over the upper surface was simulated by using a negative incidence. The results apply therefore to the equivalent of a wing at the centre of a tunnel 3 in. wide and 2·4 in. high.

When the wing incidence was changed, the recording apparatus was tilted so that the shock-waves were always recorded in planes parallel to the plane of the wing.

Some of the films were of a poor quality, partly due to the glass walls becoming dirty with oil during the test, and partly due to incorrect lighting. In the present note results from the most satisfactory sets of films are given. These cover the conditions:—

(a) $M = 1 \cdot 4$ Incidence = 4 deg.
(b) $M = 1 \cdot 6$ Incidence = 4 deg.
(c) $M = 1 \cdot 8$ Incidence = 0 deg.

The test Reynolds number, based on wing chord, was approximately $0 \cdot 25 \times 10^6$.

4. *Discussion.*—4.1. *Results.*—Figs. 5 to 10 show typical results given by the three-dimensional shock-wave recorder.

The shock-wave system produced by the wing and recorded in the present test, is made up as follows :—

(a) A leading edge shock-wave.
(b) Reflections of the leading-edge shock-wave from the surrounding tunnel walls.
(c) A shock-wave associated with the trailing edge, and the sting support.

At all Mach numbers, the leading-edge shock-wave is attached to the wing only at the centre-line. At $M = 1 \cdot 4$ the component of the free-stream velocity in a direction perpendicular to the swept-back leading edge is subsonic, and at $M = 1 \cdot 6$ it is just supersonic. At $M = 1 \cdot 8$ the component Mach number is 1·16 for which the theoretical maximum deflection for an attached shock-wave is about 3 deg and since the semi-leading-edge angle of the wing section, measured in a plane perpendicular to the leading edge is about 18 deg, the shock-wave in front of it must be detached. The observed inclination of bow-wave to direction of the free stream, measured in the plane of the wing and some distance from the centre of the wing, is given below for the various Mach numbers and wing incidences.

M	*Mach Angle* deg	*Incidence* deg	*Observed Shock-wave angle* deg
1·4	45·5	4	52
1·6	38·7	4	47
1·8	33·7	0	45

The leading-edge shock-wave is reflected at the tunnel walls. At $M = 1 \cdot 4$, the wave reflected from the floor and roof seems to form the same surface as that reflected from the side walls. At the higher Mach numbers these reflections form two separate surfaces. The reflections from the side walls however are not plotted because they were outside the region covered by the photographs.

The plottings of the shock-waves in elevation, show that at $M = 1 \cdot 4$, with the wing at 4 deg incidence (Fig. 6), there is a reflected shock-wave crossing the outboard parts of the wing. This indicates that under these conditions interference is present. For testing at this Mach number, a smaller wing would be desirable. At higher Mach numbers the reflected waves are well clear of the wing. (Fig. 8 and 10.)

The trailing-edge waves are of the same general shape as the leading-edge waves, although they seem to have a tendency to split into branches. Their shape near the centre-line is probably affected by the sting on which the model is mounted. In all cases reflections of these waves are outside the region covered in the tests.

4.2. *General Method*.—In the reduction it is assumed that differences of air density in the working-section, though they are large enough to produce shadows indicating the presence of shock-waves, are small enough for any distortion of the paths of these shadows to be neglected. The small order of this distortion can be seen in Fig. 4 by the kinks in the diagonal shadows produced by the vertical wires, where they cross the shock-wave shadows.

It has been seen that the plottings of the reflected shock-waves are incomplete, especially at the higher Mach numbers. This is due to the limited width of the band across the tunnel in which the shock-waves were recorded. For this test the band width, which is the same as the film width, was $2\frac{1}{2}$ in. The centre of rotation of this band is the origin in the shock-wave plottings (Figs. 5 to 10).

Only the leading-edge shock-wave can be plotted in the plane of the wing. Here the rest of the detail is masked by the shadow of the wing.

As this method gives only the positions of the tangents to the shock-waves a certain amount of judgment is necessary in constructing the final shock-wave patterns. This task becomes more difficult when a complicated shock-wave system is to be dealt with.

To get the complete shock-wave pattern the apparatus must be traversed so far that the band of light crossing the tunnel is at some time parallel to all the shock-waves in the plane of the band. In the present tests, this necessitated a traverse of $\pm$ 50 deg, requiring a window length about three times the width of the tunnel, with the model at its centre. At higher Mach numbers even more angular movement would be needed, though for investigating tunnel interference, the instrument is most needed at the lower Mach numbers. In larger tunnels it may be difficult to provide sufficient window space, especially behind the model. It may be more convenient therefore to apply the technique to an ancillary test in a small tunnel, such as that used in the present tests, before choosing the model size to be used in a larger tunnel.

The reduction and plotting of the photographs is rather long and tedious, in spite of having the help of the 'reduction box'. The plotting of the full shock-wave pattern for one condition of the wing, involving the reduction of 7 frames, takes one person about a day; this effort however, should be worthwhile as it would give information to enable the largest possible model to be tested in a given size of tunnel.

5. *Conclusions*.—The positions of the shock-waves in the vicinity of a 50 deg swept-back wing in a rectangular-section tunnel have been plotted by using the three-dimensional shock-wave recorder. The tests show that the outboard portions of the wing suffer interference from the tunnel walls at $M = 1 \cdot 4$. At $M = 1 \cdot 6$ and $1 \cdot 8$ the model behaves as if it were in free air.

The technique is fairly easy to apply, but as a comparatively large window is necessary, it is more suitable for smaller tunnels.

REFERENCES

No.	*Author*	*Title, etc.*
1	F. E. Lamplough	Three-dimensional shock-wave recorder. A.R.C. 11659. June, 1948.
2	W. F. Hilton	*High-speed aerodynamics*. pp. 469–70. Longmans. 1952.

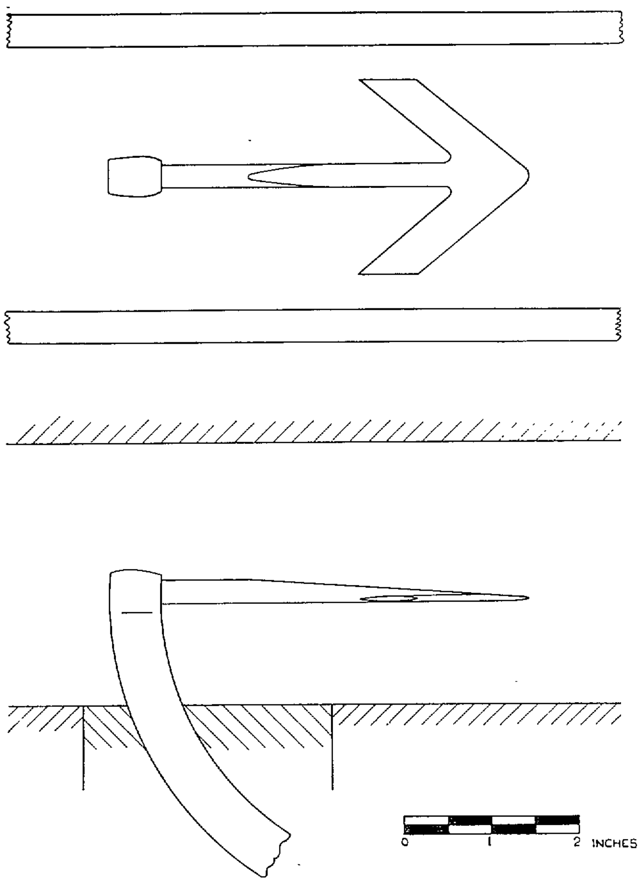


FIG. 1. General arrangement of the wing mounted in the tunnel.

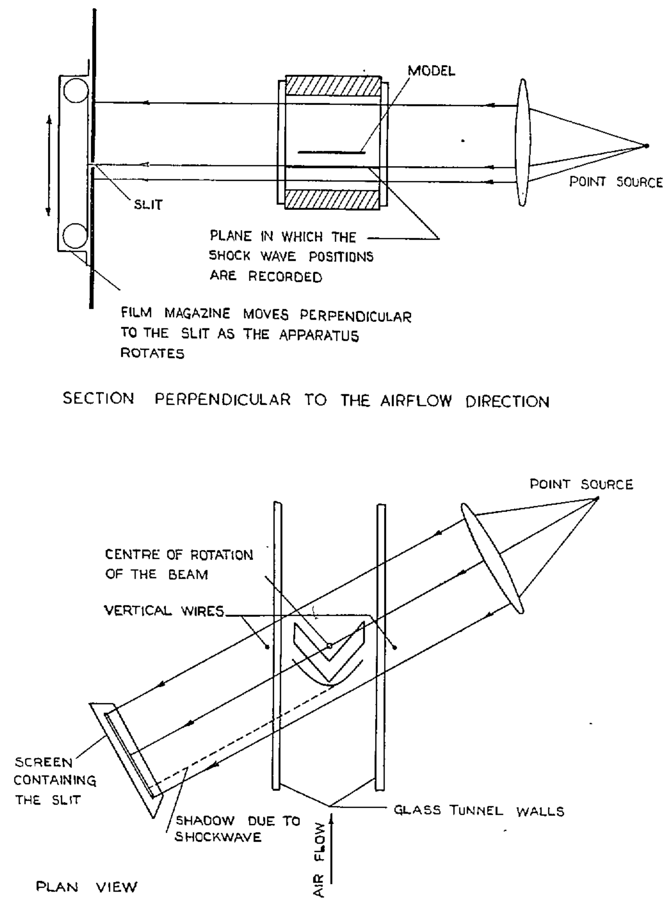


FIG. 2. The three-dimensional shock-wave recorder shown diagrammatically.

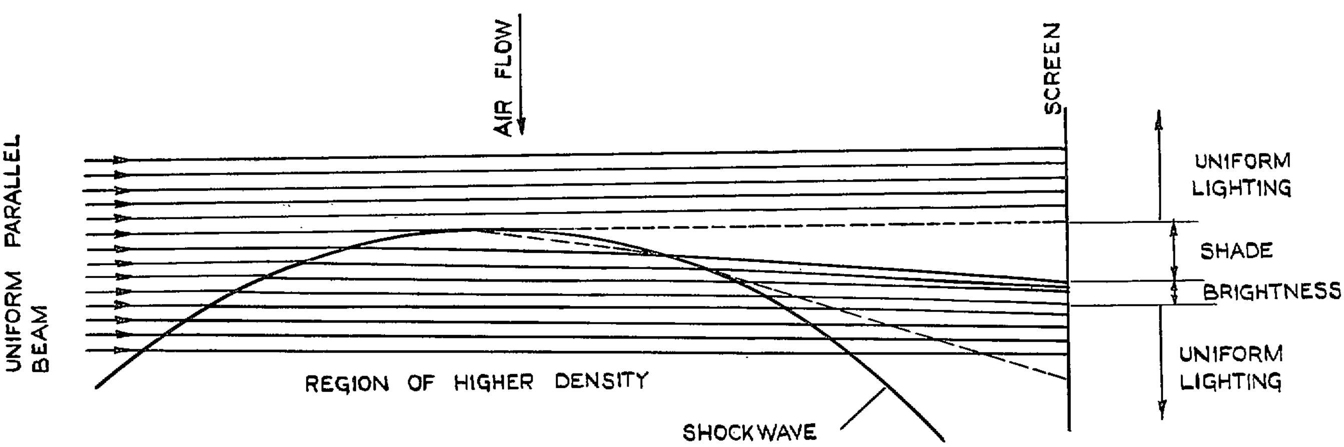
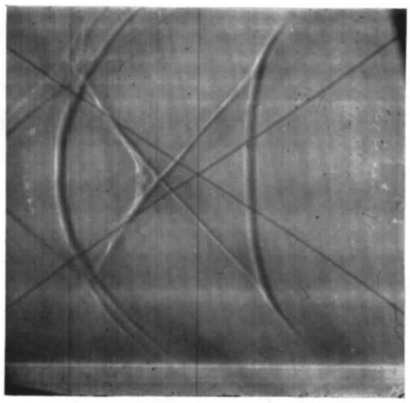
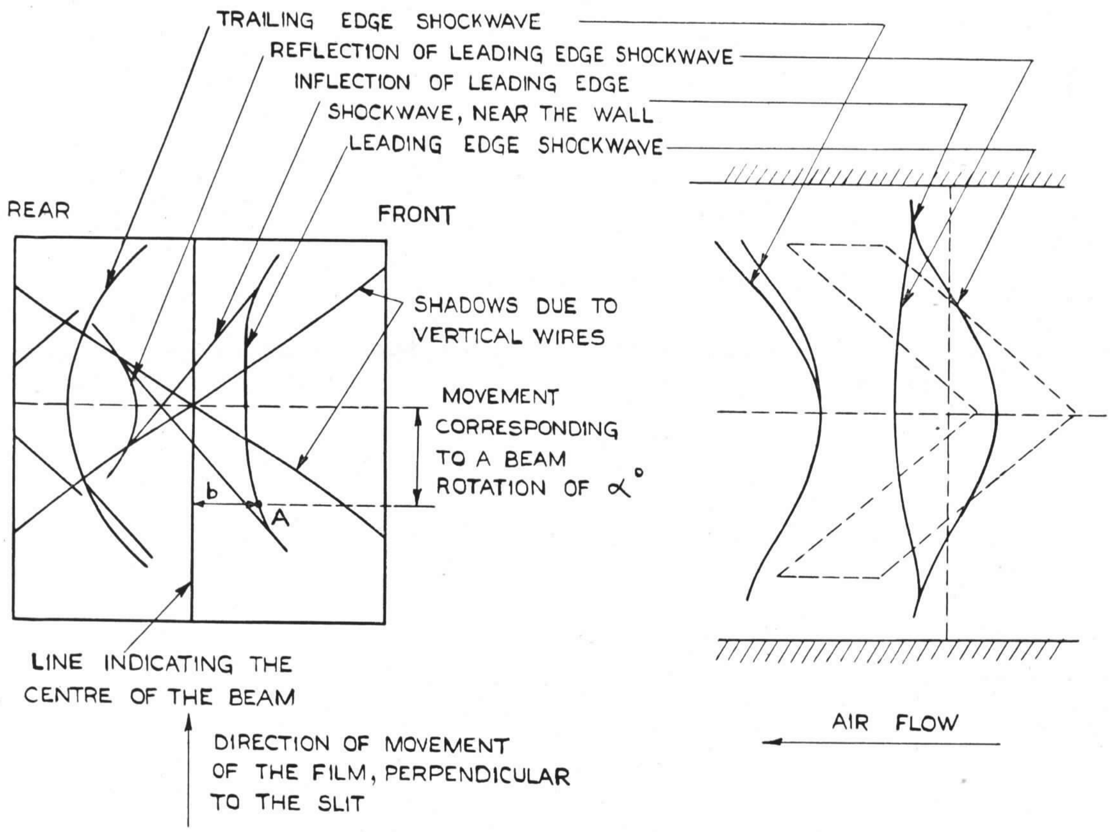


FIG. 3. Diagram illustrating the passage of a parallel beam of light through a curved shock-wave.



PHOTOGRAPH OF THE SHOCKWAVES IN A PLANE 0·66 INS. BELOW THE WING

M = 1·4, INCIDENCE = 4°



KEY TO PHOTOGRAPH

RESULTING SHOCKWAVE CONFIGURATION.

FIG. 4. Typical shock-wave photograph, with key.

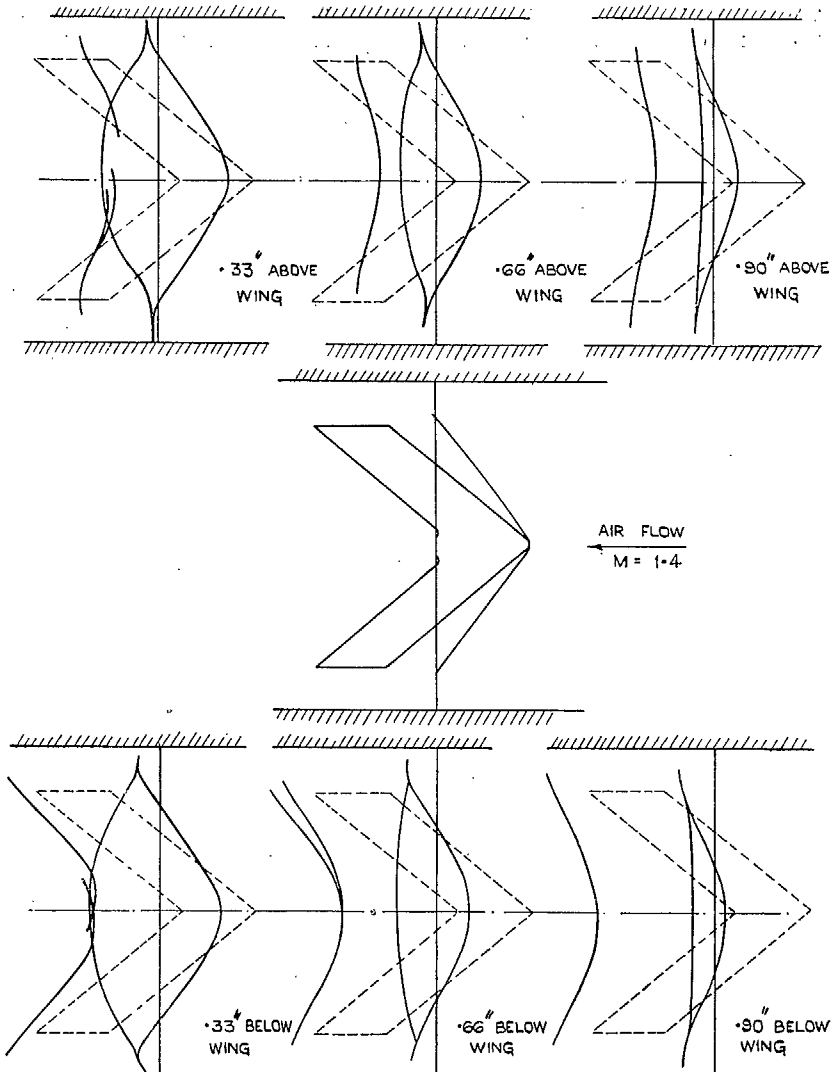


FIG. 5. Shock-wave configuration about a 50 deg swept-back wing, $M = 1 \cdot 4$. Incidence = 4 deg. Sections parallel to the plane of the wing.

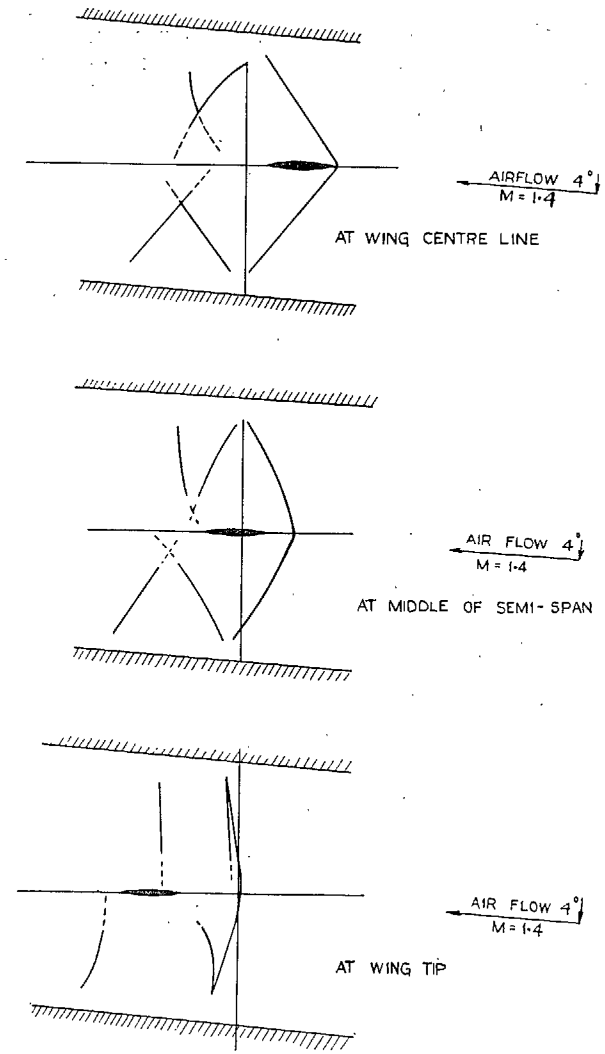


FIG. 6. Shock-wave configuration about a 50 deg swept-back wing, $M = 1 \cdot 4$. Incidence = 4 deg. Sections in line of flight, perpendicular to the plane of the wing.

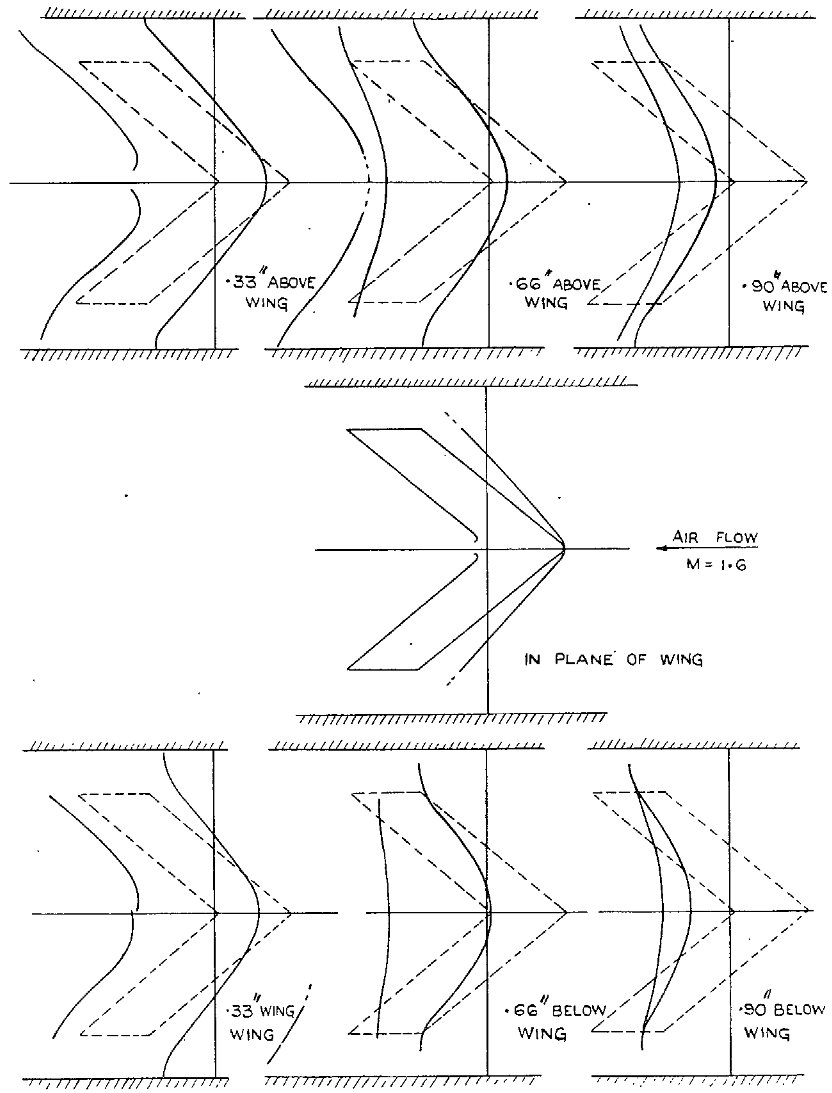


FIG. 7. Shock-wave configuration about a 50 deg swept-back wing, $M = 1\cdot6$. Incidence $= 4$ deg. Sections parallel to the plane of the wing.

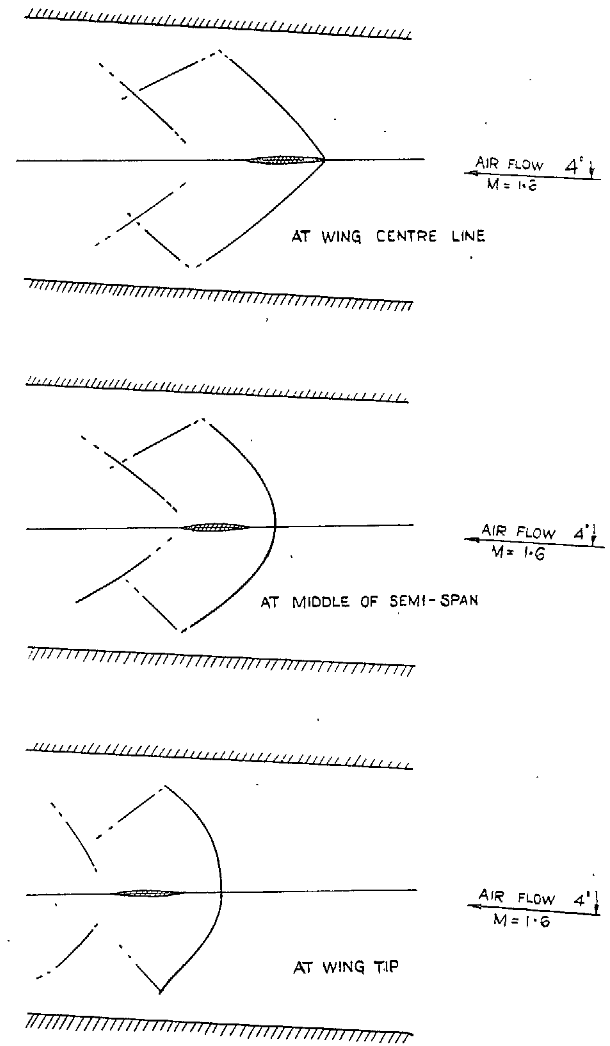


FIG. 8. Shock-wave configuration about a 50 deg swept-back wing, $M = 1\cdot6$. Incidence $= 4$ deg. Sections in line of flight, perpendicular to the plane of the wing.

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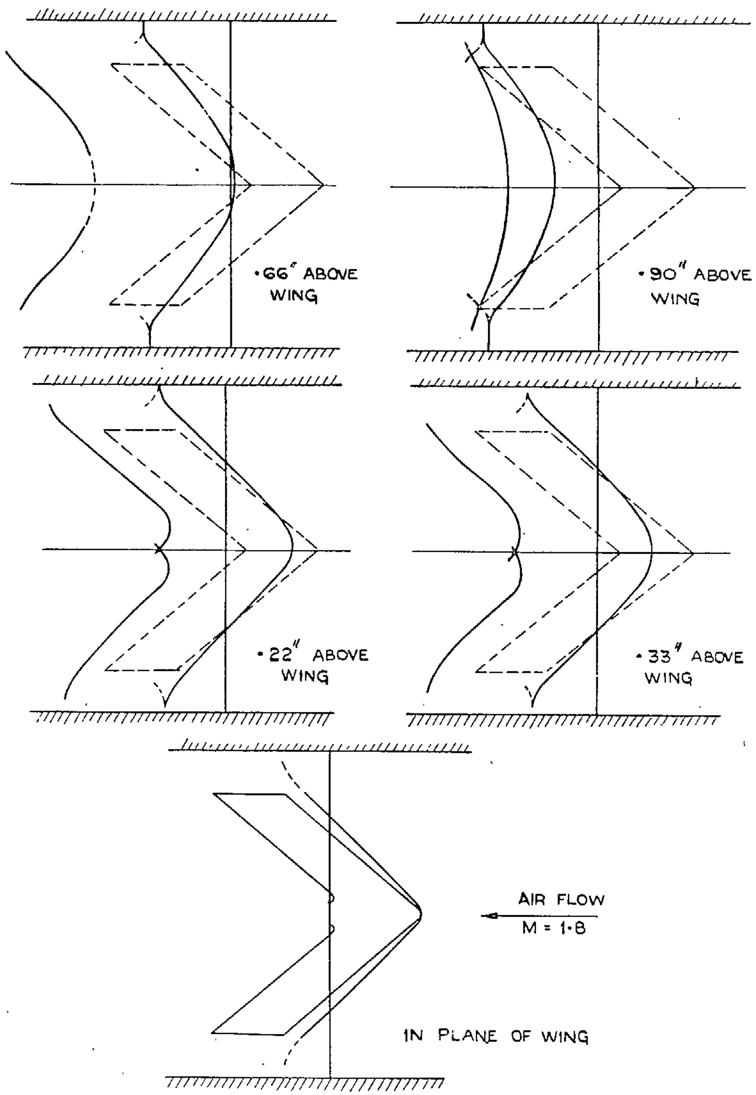

FIG. 9. Shock-wave configuration about a 50 deg swept-back wing, $M = 1 \cdot 8$. Incidence = 0 deg. Sections parallel to the plane of the wing.

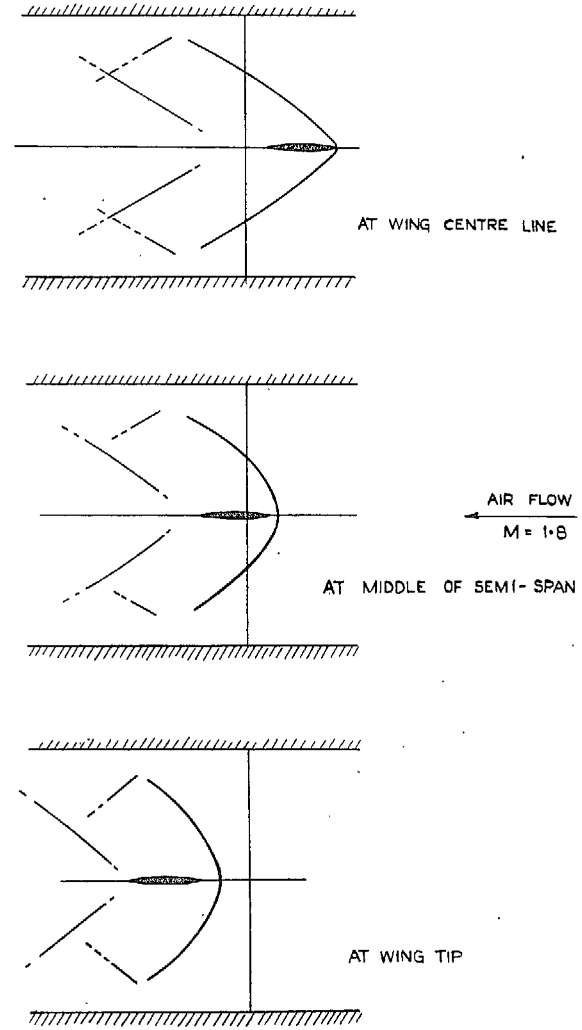

FIG. 10. Shock-wave configuration about a 50 deg. swept-back wing, $M = 1 \cdot 8$. Incidence = 0 deg. Sections in line of flight, perpendicular to the plane of the wing.

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