Experimental Investigation of the Transonic Flow at the Leeward Side of a Delta Wing at High Incidence

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An experimental investigation has been performed on the transonic flow field at the leeward side of a cropped delta wing with a sweep angle of 65°; the delta wing was mounted on a cylindrical body having an ogival nose. The experiments have been made at a free stream Mach number of 0.85 and at angles of incidence of 10° and 20°. The surface flow is visualized by oil flow pictures, showing secondary and even tertiary separations and reattachments. To determine the flow direction, the Mach number and the total- and static pressures, the flow field was explored in several cross-planes, using a 5-hole directional probe. The measurements show a complex type of flow containing dominant regions of vortices, embedded shock waves and separations. In spite of the transonic flow and the presence of a body extending in front of the wing apex a large part of the flow field may be regarded as conical. This holds mainly for the directional field and the total pressure. At a Mach number of 0.85 and an incidence of 20° a strong shock wave is observed at 80% chord position terminating a region of locally supersonic flow; the shock stands across the wing symmetry plane but its actual shape is as yet unknown. At this flow conditions also evidence is obtained of a non-conical shock between the primary vortex and the wing surface. This shock is probably generated by the upstream influence of the cropped wing tips and of the trailing edge. Both shocks interfere quite strongly with the vortex system. Furthermore some indications are found for a conical shock above the leading edge vortex.
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**NOMENCLATURE**

- $a$: speed of sound
- $c_r$: wing root chord
- $c_p$: static pressure coefficient
- $M$: Mach number
- $p$: static pressure
- $p_t$: total pressure
- $q_c$: conical velocity component, normal to a conical ray
- $q_d$: velocity component in a plane normal to the wing surface
- $q_r$: conical velocity component along a conical ray
- $u,v,w$: velocity components in $x,y,z$ directions respectively
- $x,y,z$: body oriented, righthanded cartesian coordinate system
- $\alpha$: angle of incidence
- $\Lambda$: angle of sweep ($= 65^\circ$)
- $\phi = \tan^{-1}\left(\frac{v}{u}\right) - \tan^{-1}\left(\frac{y}{x}\right)$
- $n$: conical coordinate ($= y/x$)
- $\zeta$: conical coordinate ($= z/x$)

**Subscripts:**

- $\infty$: free stream conditions
- $le$: leading edge
- $N$: normal to the leading edge
1 INTRODUCTION

The flow at the leeward side of a delta wing has a complicated structure, which is characterized by two strong and stable vortices, originating at the leading-edge. These vortices occur already at low angles of incidence, and strongly influence the aerodynamic characteristics of the wing. At greater angles of incidence also secondary, and even tertiary vortices occur as a result of boundary-layer separation. At high-subsonic and supersonic free-stream Mach numbers the flow is even more complicated due to the existence of supersonic pockets with embedded shock waves.

The vortical flow over delta wings at high angles of incidence in the low-subsonic speed range as well as in the supersonic speed range has been the subject of many experimental investigations ([1]-[6]). Investigations of the high-subsonic and transonic flow regime, however, are rare, this is in particular true for experimental results. The computation of the vortical flow field past a delta wing in all its aspects, such as rotational and viscosity effects, is a very difficult task, and experimental results in the high-subsonic and transonic flow regime are very welcome indeed.

The purpose of the present report is to deliver some fundamental results of the vortex flow above a delta wing with 65° sweep angle; the wing has a flat upper surface and sharp leading edges. Emphasis will be put on phenomena occurring at angles of incidence of 10° and 20° for a free stream Mach number of 0.85; the results include surface oil flow patterns and detailed flow field explorations.
2 EXPERIMENTAL APPARATUS AND TESTS

2.1 Wind tunnel

The experiments have been performed in the TST-27 transonic-supersonic wind tunnel of the Faculty of Aerospace Engineering at the Delft University of Technology. The wind tunnel is of the blow-down type and has a closed test-section with a cross-section of 28x25.3 cm². It is equipped with a variable nozzle and can be operated in the Mach number range from M=0.5 to M=5.5. To operate the wind tunnel at high subsonic Mach numbers, the variable nozzle is completely opened and the Mach number in the test section is controlled by an adjustable choke, downstream of the test section. The Reynolds number based on the wing root chord, can be varied from $1.8 \times 10^6$ - $5.6 \times 10^6$ by variation of the stagnation pressure in the wind tunnel. Most tests were performed at a Reynolds number of $3.5 \times 10^6$.

2.2 Experimental set-up

The model geometry is shown in fig. 2.1. The cropped delta wing with a flat upper surface and a leading edge sweep $\alpha = 65^\circ$ is mounted on a cylindrical fuselage, extending in front of the wing; the fuselage had an ogival nose. The planform of the wing and fuselage is based on an early design concept of the model proposed by the Joint US/European Transonic Vortex Flow Experiment Group.

The experimental set-up is sketched in fig. 2.2. For the flow-field explorations a 5-hole directional probe was used, which was mounted on a sting. The probe could be traversed in all directions, thus covering completely the part of the test-section occupied by the delta wing. The probe was aligned along the centre-line of the test section. During the measurements the probe was traversed in a plane normal to the free-stream vector. The probe had a spherical head and its outer diameter was 1.6 mm (fig. 2.3). The head was provided with five static pressure orifices, one at the centre and four circumferentially every 90 degrees, under 45 degrees with the centre-line; the diameter of the orifices was 0.2 mm.

An extensive calibration of the probe provides the possibility to determine all local flow quantities. The necessary local flow quantities, which are flow
direction, Mach number and total pressure, are related to the probe pressures using a number of independent calibration coefficients. For a description of the calibration and processing of measurements, see [7]. The probe was calibrated for several Mach numbers between 0.55 and 1.8 and for angles of pitch up to 60 degrees. The calibration was made for sets of the angle of pitch and the angle of roll of the probe; for a discrete angle of pitch the angle of roll was varied.

A righthanded wing oriented cartesian coordinate system \((x,y,z)\) is defined, with the \(x\)-axis along the wing root chord and the \(z\)-axis perpendicular to the wing; the origin is located at the apex of the wing.

2.3 Measurements

Flow visualization studies were made using surface oil flow techniques and a conventional color Schlieren system. The oil mixture for the surface flow pictures consisted of Tellus 29 oil and titanium dioxide in a composition of 3 to 2; as a detergent a few droplets of oil acid were added.

For a good contrast the model was painted black; the extending nose of the body is not visible on the pictures. These tests were performed by J.A. Möller [14].

The flow field measurements were performed at two different angles of incidence, \(\alpha = 10^\circ\) and \(20^\circ\), at a free-stream Mach number \(M_\infty = 0.85\). The measurements were made in cross-flow planes normal to the free-stream direction by traversing the probe in spanwise direction at constant heights above the wing surface. In order to find out the size and shape of the supersonic pocket and shock wave in the symmetry plane of the wing also measurements were made in that plane at fixed streamwise positions, but with the probe moving in upward direction (away from the wing surface).

At each measuring point the pressure measurements were taken in a stationary position. The probe measurements were corrected for pressure gradients.
3 CONICAL FLOW PROPERTIES

The geometry of the delta wing may give reason to the assumption of conical flow. In general this is justified in the supersonic case, where there is practically no upstream influence, but in the subsonic- and transonic speed range where trailing edge influences exist, such an assumption need not be valid. However, the present and other investigations (see [5], [6], [14]) give indications about at least a geometrical conical flow. By this we mean a flow wherein the local flow angles are constant along straight lines originating in the apex (or a point close to it). It implies also that certain phenomena like flow separation from and reattachment to the wing surface, the core of the vortex, occur along the conical rays.

Usually the directional field in a cross-flow plane perpendicular to the wing root chord is represented in a non-conical way, which means that the directional field is composed of the cross-flow velocity components \( v \) and \( w \) (see fig. 3.1). This representation however does not show the correct positions of separation- and attachment lines and vortex cores, since these lines are conical rays and the direction of the velocity vector is along them. A conical representation of the directional field will show such lines as conical stagnation points in the cross-flow-plane. In real conical flow (the flow quantities are constant along rays through the apex) conical streamsurfaces may be formed. The intersection of these surfaces with the cross-flow-plane gives the conical streamlines, of which the local direction is given by the components \( v-u_\eta \) and \( w-u_\zeta \) (see fig. 3.2). With this representation the position of the vortex core is mapped into a point in the cross-flow-plane where \( v-u_\eta = 0 \) and \( w-u_\zeta = 0 \), and separation- and attachment lines into a point where \( \zeta = 0, w = v-u_\eta = 0 \).

The magnitude of the conical velocity component \( q_c \), which is the velocity component normal to a conical ray, is defined by:

\[
q_c = \frac{1}{\sqrt{1+\eta^2+\zeta^2}} \cdot \sqrt{(v-u_\eta)^2 + (w-u_\zeta)^2 + (v_\zeta - w_\eta)^2}
\]  

(3.1)

It gives some indication about the existence of conical shock waves, since the conical velocity in front of a conical shock must be supersonic, i.e. the conical Mach number \( M_c = q_c/a > 1 \); downstream of the shock wave the conical Mach number decreases.
4 PHYSICS OF THE FLOW AROUND A DELTA WING

At angles of incidence of about 4° and higher the flow at the leeward side of a wing with a delta planform is characterized by large separated flow areas. This separated flow pattern, however, retains a highly stable structure. In the flow pattern two leading edge vortices occur, as shown in fig. 4.1. The flow reattaches at the upper surface at the so-called primary attachment lines (A1). Between the primary vortices there exists a large area of flow which does not merge into the primary vortices. At greater angles of incidence also secondary vortices may arise (see fig. 4.1, S1), due to boundary-layer separation. This separation is caused by a strong pressure gradient on the wing surface from the primary attachment line towards the leading edge. Two dimensional sketches are given in figs. 4.2 and 4.4 according to a conical representation. The primary attachment lines move towards the wing symmetry line as the angle of incidence increases, and at a given angle they coincide at the wing symmetry line. This structure is shown in fig. 4.3 and 4.4.

The physical picture of the vortical flow at low speeds is given in fig. 4.5 (see [8]). At the leading edge a free shear-layer is formed by the boundary-layers of the upper- and the underside of the wing. Characteristic of this shear-layer is a jump of the magnitude and direction of speed through the shear-layer. Besides that there is a loss of total pressure in the shear-layer. As the layer curls up its thickness is growing due to the difference in speed at both sides. Eventually a rotational core is formed with a so-called inner viscous sub-core where large gradients of velocity and pressure are present.

In the vortex flow the velocity differs much from the external flow. For low subsonic flows the axial velocity along the vortex core can reach values up to 3 times the free-stream velocity ([9] and [10]). This means that at low free-stream Mach numbers (\( M_a > 0.5 \)) local supersonic pockets can occur where the flow structure can be complicated due to embedded shock waves.

The experiments of Vorropoulos and Wendt ([10] and [11]) give some information about the vortical flow field at a high-subsonic free-stream Mach number. Their LDV measurements, carried out at a flat delta wing with 62° sweep and an angle of incidence of 10°, and at a free-stream Mach number of 0.8 show a decrease of the axial velocity (along the vortex axis) in the neighbourhood of the vortex
core. They also measured in a horizontal traverse through the vortex core a asymmetrical distribution of the vertical velocity w. Vorropoulos and Wendt suppose that this is caused by a conical shock wave between the primary vortices and the wing surface. In their experiments the existence of this shock, however, was not be demonstrated by pressure measurements or confirmed by discontinuities in the velocity distribution.

Whether the flow structure with leading edge vortices is maintained also in a supersonic flow depends on the Mach number and angle of incidence normal to the leading edge, given by

\[ M_N = M_\infty \sqrt{1 - \sin^2 \Lambda \cos^2 \alpha} \]

(4.1)

and

\[ \alpha_N = \tan^{-1} \left( \frac{\tan \alpha}{\cos \Lambda} \right) \]

(4.2)

respectively. If the normal Mach number reaches supersonic values, an expansion flow without separation may arise. The different flow structures that are possible are classified in the well-known \( \alpha_N-M_N \) diagram (fig.4.6) taken from [13].

In this diagram also the positions of the present measurements at \( M_\infty = 0.85 \) and \( \alpha = 10^\circ \) (1) and \( 20^\circ \) (2) are indicated.

In some experiments on delta wings at high subsonic free-stream flows a shock was detected just upstream of the trailing edge, see [14], [15] and [16]. This shock, which terminates a supersonic pocket, is observed in schlieren pictures. The experiments of Sutton [15] show that the terminating shock exists over the whole span. With increasing free-stream Mach number the shock moves in a downstream direction, whereas with increasing angle of incidence it moves upstream. Lambourne and Bryer [16] believe this terminating shock be a possible cause of vortex-bursting, because on schlieren pictures it is seen as a turbulent continuation of the vortex behind the shock.
5 DISCUSSION OF RESULTS

The present delta wing configuration has been subject to extensive surface oil flow visualization tests at various angles of incidence and free stream Mach numbers [14]. Some of the results at a free stream Mach number of 0.85 and at angles of incidence of 10°, 15° and 20° are shown in the figures 5.1 - 5.3. These pictures reveal clearly the primary attachment and secondary separation lines. At an angle of incidence of 10° a rather large interior region exists, where the oil streaklines are directed more or less parallel to the symmetry line of the wing. At higher angles of incidence (15° and 20°) the primary attachment lines have moved towards the wing symmetry line and coincide with it. At these angles of incidence also tertiary separation lines exist. Near the trailing edge these lines end in foci, indicating that the flow leaves the surface in discrete spirals. The straight attachment and separation lines over a great part of the wing suggest that the flow over the wing may be regarded as conical, as far as the directional field is concerned. The attachment and separation lines intersect in front of the apex. This may be due to the body, which is not visible on the pictures. It is not clear whether we are dealing with laminar or turbulent separation. If transition takes place somewhere on the wing, a kink in the secondary separation line should be observed. Taking into account a Reynolds number of 5.6x10⁴ one might expect a turbulent boundary layer, therefore the existence of the body extension in front of the wing apex may give a premature transition.

In fig. 5.4 the angle between the streaklines and the rays from the apex and defined by \( \varphi = \tan^{-1}(v/u) - \tan^{-1}(y/x) \) is drawn for 4 angles of incidence at \( M_s = 0.85 \). This figure reveals clearly the coincidence of the primary attachment line with the wing symmetry line at angles of incidence of 15° and 20°. Also the secondary separation lines move towards the wing symmetry line with increasing angle of incidence. Prior to separation the streaklines show a strong curvature towards the separation line.

In the fig. 5.5 the direction of the oil streaklines is compared with the flow directions measured with the 5-hole probe in two traverses just above the surface. At 3=10° (fig. 5.5.a) the direction of the oil streaklines and the flow near the surface is more or less the same between the primary attachment lines and the position of the primary attachment line is well predicted by the probe
measurements. Between the primary attachment line and the secondary separation line the oil streaklines are stronger curved towards the leading edge than the flow streamlines (fig. 5.5.a and b). This is a well-known phenomenon from boundary layer calculations [17].

Next we consider the results of the flow field explorations to start with α=10°. In the figures 5.6 and 5.7 the directional field in the cross-flow-planes \( x/c_r = 0.6 \) and 0.8 is shown in conical representation. The positions of the primary attachment line \( (A_1) \) and the secondary separation line \( (S_2) \) have been obtained from the oil flow visualization tests [14]. The directional field suggests an attachment line, almost at the same place as has been found with the oil flow visualization tests (see also fig. 5.5a). The conical directional field is different from a directional field composed of the velocity components \( v \) and \( w \), see fig. 5.8, which does not show a point of primary attachment. As has been discussed in section 3, a conical representation is preferable in this case. In the figs. 5.6 and 5.7 also the free stream total pressure contour \( (p_t/p_t e = 1) \) and the sonic line are drawn. The \( p_t/p_t e = 1 \) line encompasses a region of flow that is strongly affected by the shear flow around the leading edge. The isobars of the total pressure in the cross-flow-plane \( x/c_r = 0.6 \) are drawn in fig. 5.9, which shows the great influence of the shear flow of the leading edge vortex. In the primary vortex the total pressure is decreased to 60% of the free stream value.

The static pressure, calculated from the total pressure and the Mach number, is given as isobars of the pressure coefficient \( c_p \) in fig. 5.10. In the region between the symmetry plane and the primary attachment line the \( c_p \)-value is nearly constant, in this region the Mach number is almost 1 \( (c_p = 0.3029) \). The pressure minimum lies near the vortex core, which is to be expected because of the curvation of the streamlines around the vortex core. Remarkable in fig. 5.10 is a region with a relative high pressure \( (c_p = 0) \) near the wing surface at a span wise position \( y/y_{le} = 0.5 \). To illustrate this the results of two probe traverses near the surface are plotted in fig. 5.11. The pressure distribution above the wing surface shows a relative maximum at \( y/y_{le} = 0.5 \), while the pressure level is decreasing under the primary vortex. Such a pronounced local pressure maximum at this place is not measured at the surface, see e.g. the experiments of Muylaert [11]. The measurements of the flow direction, however, agree well with the observations in the oilflow pictures, as has
been shown in fig. 5.5a, where the angle $\phi$ of the oilflow streaklines is compared with the angle in the probe traverse nearest to the surface.

Lines with a constant conical Mach number $M_c$ are shown in fig. 5.12. Nowhere the conical Mach number exceeds unity. The highest values of $M_c$ are reached just above the primary vortex core ($M_c = .95$) and between the primary and secondary vortices ($M_c = .9$). At this place it is possible that a conical shock wave appears when the Mach number in the flow field increases and the conical Mach number reaches supersonic values as a result of an increased free stream Mach number and/or angle of incidence.

The present results are different from the observations of Vorropoulos and Wendt [11], who have done LDV-measurements in the flow above a delta wing model under comparable circumstances. They find some indications for a conical shock wave between the primary vortex and the wing surface. Their hypothesis for the existence of such a shock is based on the decrease of the axial velocity component in the primary vortex core and on the decrease of the velocity component normal to the wing surface in the field between the vortex core and the shear-layer from the leading edge. To illustrate the behaviour of the axial velocity component along a conical ray $q_T$, is shown in fig. 5.13 for some horizontal traverses through the vortex core. It turns out that $q_T$ in the vortex core is higher than outside it. To compare this with the measurements of Vorropoulos and Wendt, it should be noted that the LDV - measurements provide velocities, while the present experiments give Mach numbers. For constant total temperature Mach numbers can be deduced into velocities and a qualitative comparison can be made, this is how fig. 5.13 originated. Then the results of the present experiments show a different behaviour for the axial velocities than the results of [11], also the maximum axial velocity is higher $1.5 \times U_\infty$ as compared to $1.12 \times U_\infty$ in [11].

The measurements in the wing symmetry plane are given in fig. 5.14, where the isobars of the static pressure coefficient are drawn. Upstream of the station $x/c_T = 0.6$ the flow seems rather conical; downstream of this station the influence of the trailing edge is clearly noticeable because the lines of constant $c_p$ are almost normal to the wing surface. Upstream of $x/c_T = 0.65$ in a region close to the surface the critical $c_p$ value (-0.302) is exceeded.
Increasing the angle of incidence from 10° to 20° at a free stream Mach number of 0.85 results in some significant changes. The oil flow visualization shows the coincidence of the primary attachment lines with the symmetry line. Furthermore a shock wave is observed in the schlieren pictures. Fig. 5.15 shows a schlieren picture, taken at the conditions \( M_\infty = 0.85 \) and \( \alpha = 20° \), and fig. 5.16 gives the interpretation of this picture. The dimensions of the supersonic pocket and the position of the sonic line has been estimated from wavelets on the photograph. These wavelets are caused by irregularities in the surface at the symmetry line. The position of the vortex core is represented as a narrow wedge of a light color. The shock wave appears rather thick in the picture, which, apart from small instabilities, is due to a curvature of the shock in spanwise direction. From the present probe measurements only suggestions could be obtained for the shape of the shock. It should be noted, however, that in the oil flow visualizations this shock was not observed.

The probe measurements in the wing symmetry plane confirm the existence of the shock, which can be seen in fig. 5.17, where the isobars of the static pressure coefficient are drawn. The clustering of the isobars at \( x/c_{\tau} = 0.8 \) suggests a shock wave. Since there are no total pressure losses in the wing symmetry plane in front of the shock the position of the sonic line coincides with the isobar with the value of the critical pressure coefficient \( \gamma = -0.302 \). This isobar is also not very different from the sonic line, estimated by the schlieren picture (fig. 5.16). In front of the shock the isobars above the level \( \gamma = -0.55 \) have a rather conical shape; closer to the surface isolated regions with lower pressures occur.

Of all measurements in the cross-flow planes \( x/c_{\tau} = 0.3, 0.6, 0.8 \) and 1.0, those made at station \( x/c_{\tau} = 0.6 \) are the most detailed. The directional field is drawn in conical representation in fig. 5.18; the figure shows the coincidence of the primary attachment lines with the wing symmetry line. Near the surface between \( y/y_{fe} = 0.75 \) and 1.0 the influence of the secondary vortex is noticeable. In this figure also the free stream total pressure contour \( (p_t/p_{t_{\infty}} = 1) \) and the sonic line are drawn. The free stream total pressure line \( (p_t/p_{t_{\infty}} = 1) \) encompasses a region of flow that is strongly affected by the shear flow around the leading edge.

The directional field in this figure is different from that of the cartesian representation shown in fig. 5.19. The position of the vortex core is estimated from the point with a minimum in total pressure, which gives a more reliable
indication for this position than the directional field does because the latter is strongly influenced by the way of presentation.

The total pressure contours given in fig. 5.20, show that in the core of the primary vortex the total pressure is less than 40% of its free stream value. After one turn around the vortex core part of the outer flow passes between the primary and secondary vortices, as can be seen from a region with a relative high total pressure. The contraction in the $p_t$-contours may be explained by the fact that the just mentioned high total pressure flow interferes with the inner part of the shear layer coming from the leading edge, thus causing a recovery of the total pressure there.

At the trailing edge station $x/c_r = 1.0$ the region with a very low $p_t$ is much larger than at the station $x/c_r = 0.6$, as shows fig. 5.21. This reveals the consequence of the interaction between the terminating shock at $x/c_r = 0.8$ and the vortex, which probably leads to vortex bursting. In the region where $p_t/p_{t\infty} = 0.3$ all pressures measured by the 5 orifices of the probe are nearly equal, indicating a very low velocity or even backward flow. The schlieren pictures show a blurred continuation of the vortex behind the shock, also suggesting vortex bursting. Probe effects have to be discarded since the schlieren pictures do not show any difference between runs with and without probe.

The isobars of the static pressure coefficient at the station $x/c_r = 0.6$ (fig. 5.22) show above the primary vortex core, where high supersonic Mach numbers have been measured, a pressure minimum. However, there is no acceptable explanation for the different positions of the vortex core and the pressure minimum. Probably, considering the probe size (about 0.022 in the units of fig. 5.22), the determination of the Mach number is inaccurate in the vicinity of the core.

A marked phenomenon is the strong pressure gradient underneath the vortex, at $y/y_{le} = 0.5$, which is illustrated by fig. 5.23 for two traverses underneath the vortex. The pressure gradient is adverse since the flow is directioned from the symmetry plane towards the leading edge. Such a behaviour was also observed at $\alpha = 10^\circ$ (fig. 5.11), but is much stronger here for $\alpha = 20^\circ$. So it appears that we have to do with a structural phenomenon that is intensified as the angle of incidence increases. Through the almost discontinuous pressure rise the Mach number decreases from supersonic (1.3) to subsonic (0.85) values.

Because of these observations one might decide for a shock at this place. According to the present measurements it does not seem to be a conical shock. In
the case of a conical shock, an oncoming flow directed away from the symmetry plane would bend towards the symmetry plane, when crossing the shock. The present measurements do not reveal such a behaviour. As is illustrated in fig. 5.24 the ratio of $M_{d}$ (derived from $\sqrt{u^2 + v^2}$) and $M_{x}$ (derived from $u$) does not increase at the shock position, but it rather increases. If this is taken into account, one should decide for a shock running like the line PQ in fig. 5.25. This tentative conclusion is also supported by preliminary measurements [14], showing an analogous behaviour at $x/c_{r} = 0.67$ (point Q). The surface streaklines have been traced from the oil flow pattern, which agrees well with the directional field near the wing surface (see fig. 5.5b). It is by no means clear, whether this shock is connected with the shock wave across the symmetry plane (also indicated in fig. 5.25), and if so, how this connection takes place.

The accuracy of the probe measurements near the surface might be questionable; the measured increase of the pressure in the flow field close to the wing surface should also be measured on the surface, however, for the present model and flow conditions this kind of measurements are not yet available. Experiments with delta wings under comparable circumstances do not show sharp pressure increases on the surface.

The contours of the conical Mach number $M_{c}$ in the cross-flow-planes $x/c_{r} = 0.3$ and 0.6 are shown in figures 5.26 and 5.27. In both planes the conical Mach number reaches supersonic values above the vortex core. At a spanwise position $y/y_{h} = 0.6$ between the heights $z/y_{h} = 0.25$ and 0.45, the conical Mach number decreases in flow direction. This is accompanied by a decrease in $p_{t}$, which suggests the existence of a weak oblique conical shock wave. The measured changes are not sharp enough to make sure about a shock wave, since the quantities could be smeared out over several stations, due to the bow shock wave of the probe.

In the cross-flow-plane $x/c_{r} = 0.3$ also supersonic values of $M_{c}$ are reached between the primary and secondary vortex. One condition for the existence of a conical shock wave at this point, coinciding with the position of the secondary separation, is fulfilled; also at this position the flow bends towards the symmetry plane. However, apart from supersonic values of $M_{c}$ there are no further indications whether a weak shock wave at this place could be present.
6 CONCLUDING REMARKS

The compressible flow at the leeward side of a delta wing at high subsonic and transonic speeds is a very complicated subject, experimentally as well as theoretically. The probe measurements give a good insight in the geometrical structure of the flow field (flow direction and total pressure), but interesting details, such as the shape of the supersonic pocket, the position and shape of embedded shock waves and the behaviour of vortices due to interaction with shock waves, are as yet not clarified. Only some suggestions can be made from the results of this paper.

The major conclusions, some of them tentative, may be formulated as follows:

- In principle the flow is not necessarily conical in the strict sense, that means flow properties need not to be constant along rays through the wing apex. However, geometrically speaking, certain important phenomena occur along straight lines through the wing apex (e.g. separation, reattachment, vortex cores) or take place more or less inside a conical surface (e.g. boundaries of vortex induced regions).

- Large parts of the flow field in the exterior region of the primary vortex are supersonic.

- The loss in total pressure due to the vortex system is considerable. In the inner region of the vortex it may rise to 35% and 60% of the free stream value at $\alpha = 10^\circ$ and $20^\circ$ respectively at a free stream Mach number of 0.85.

- At $M_\infty = 0.85$ and $\alpha = 10^\circ$ no evidence is found for embedded shock waves, in spite of the high Mach numbers measured.

For $M_\infty = 0.85$ and $\alpha = 20^\circ$ the following conclusions may be added:

- At a streamwise station ($x/c_r = 0.8$) a shock wave terminating a supersonic region was measured in the symmetry plane of the wing. The shape of this shock in spanwise direction is as yet not clear. Behind this shock the flow may not be regarded as conical in the geometrical sense anymore.

- At ($x/c_r = 0.6$) and downstream of it a shock wave was detected between the primary vortex and the wing surface. This shock does not seem to be of a conical
nature, since its orientation is opposite to rays through the wing apex. The shock is probably due to the influence of the trailing edge and the cropped wing tips.

- Above the vortex core at the stations \(x/c_p = 0.3\) and \(0.6\) some indication is found for the existence of a conical shaped embedded shock wave.

- Downstream of the terminating shock across the wing symmetry plane the primary vortex changes its character. It is bend away from the wing surface and towards the symmetry plane. At the trailing edge the core of the primary vortex is very wide; this phenomenon possibly precedes vortex bursting.
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Fig. 2.1: Delta wing model
Fig. 2.2: Experimental set-up
Fig. 2.3: Five-hole directional probe
Fig. 3.1: Velocity components with respect to a cartesian coordinate system.

Fig. 3.2: Velocity components used in conical flow representation.
Fig. 4.1: Vortical flow pattern above a delta wing, moderate angle of incidence

Fig. 4.2: Conical streamlines above a delta wing, moderate angle of incidence
Fig. 4.3: Vortical flow pattern above a delta wing, high angle of incidence

Fig. 4.4: Conical streamlines above a delta wing, high angle of incidence
a: SHEAR-LAYER  
b: ROTATIONAL CORE  
c: VISCOUS SUB-CORE  
d: SECONDARY VORTEX

Fig. 4.5: Physical flow picture of the flow above a delta wing at low speed

Fig. 4.6: Diagram showing the various delta wing flow scenarios (Ref. 13)
Fig. 5.1: Surface oil flow pattern;
$M_\infty = 0.85$, $\alpha = 10^\circ$, $Re = 5.6 \times 10^4$

Fig. 5.2: Surface oil flow pattern;
$M_\infty = 0.85$, $\alpha = 15^\circ$, $Re = 5.6 \times 10^4$

Fig. 5.3: Surface oil flow pattern;
$M_\infty = 0.85$, $\alpha = 20^\circ$, $Re = 5.6 \times 10^4$
Fig. 5.4: Distribution of angle between the oil streaklines and the conical rays
Fig. 5.5: Distribution of the angle of streamlines close to the surface, compared with the angle of the oil streaklines.
Fig. 5.6: Conical flow directions ($x/c_r=0.6$)

Fig. 5.7: Conical flow directions ($x/c_r=0.8$)
Fig. 5.8: Cross-flow directions (v, w field)

Fig. 5.9: Lines of constant total pressure
Fig. 5.10: Lines of constant pressure coefficient \( c_p \)

Fig. 5.11: \( c_p \)-distributions in traverses near the surface
Fig. 5.12: Lines of constant conical Mach number

Fig. 5.13: Distribution of axial velocity near the core of the primary vortex
Fig. 5.14: Lines of constant pressure coefficient $c_p$ in the symmetry plane
Fig. 5.15: Schlieren picture at $M_\infty=0.85$ and $\alpha=20^\circ$

Fig. 5.16: Interpretation of the schlieren picture of fig. 5.15
Fig. 5.17: Lines of constant pressure coefficient $c_p$ in the symmetry plane.
Fig. 5.18: Conical flow directions

Fig. 5.19: Cross-flow directions (v, w field)
Fig. 5.20: Lines of constant total pressure

Fig. 5.21: Lines of constant total pressure in the trailing edge cross-plane
Fig. 5.22: Lines of constant pressure coefficient $c_p$

Fig. 5.23: $c_p$-distributions near the wing surface
Fig. 5.24: Distribution of flow deviation

Fig. 5.25: Suggested plan view of surface flow with shocks
Fig. 5.26: Lines of constant conical Mach number,
\( x/c_r = 0.3 \)

Fig. 5.27: Lines of constant conical Mach number,
\( x/c_r = 0.6 \)