AN EXPERIMENTAL INVESTIGATION OF THE INTERACTION BETWEEN A SHOCK WAVE AND A TURBULENT BOUNDARY LAYER ON A CONVEX WALL

C. Nebbeling

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Acknowledgement

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Summary

The interaction between a shock wave and a turbulent boundary layer on the convex wall of a curved wind tunnel canal has been investigated. The radius of curvature, the boundary layer thickness and the Mach number were chosen so that a closed separated region near the shock wave was obtained. By means of an adjustable choke far downstream of the interaction region the maximum value of the Mach number just upstream of the shock wave was kept at $M = 1.43$. The thickness of the undisturbed turbulent boundary layer was $6.2 \text{ mm}$ and the Reynolds number $Re_\delta$ related to this boundary layer thickness was about $2 \times 10^5$.

The flowfield has been investigated by means of static pressure probes, pitot probes and a preston probe. From these measurements Mach number and velocity distributions have been deduced as well as the boundary layer integral parameters and the skin-friction coefficients. The length of the separated region, related to the undisturbed boundary layer thickness was smaller than usually found on a plane wall.

The influence of wall curvature on the boundary layer integral parameters appears from comparison with issued results obtained from plane wall shock wave-boundary layer interactions.
1. Introduction

As during the last decennia the speed of civil aircraft increased up to transonic velocities, the aims on wing design increased as well. Particularly at the transonic flow regime a lot of phenomena appeared which are unknown at subsonic flight. Most obvious is the presence of normal shock waves in the flowfield of the wing. Apart from their contribution to the drag of an airplane owing to wave drag, these shock waves interact with the wing boundary layer causing a decrease of the lift over drag ratio.

Local separation of the boundary layer near the foot of the shock wave may deteriorate the efficiency of the wing even more and still worse will be a complete separation of the flow up to the trailing edge of the wing.

Although Ackeret, Feldmann and Rott (Ref. 1) made their extensive experimental investigations on shock wave-boundary layer interaction in a curved flow as early as 1946, most research up to now on this subject has been made in a straight parallel supersonic flow along plane walls (Ref. 2, 3, 4, 5, 6). In general the latter type of flow is rather stable and without the problems of instabilities as they occur in a mixed transonic flow with shock waves. These instabilities of a curved flow make high demands on research technique. Arguments for bringing this effort on oneself appeared not clearly until 1973. In that year Bradshaw (Ref. 7) called attention to the influence of streamline curvature on the eddy viscosity in turbulent boundary layers. Based on that influence Inger (Ref. 8) made engineering approximations to correct the skin-friction coefficient and the incompressible shape factor for small longitudinal curvature and non-separating flow. His calculations lead to substantial influence of streamline curvature on both quantities. This underlined the necessity to involve wall curvature in the investigation of shock wave-boundary layer interaction in case it concerns a study to support the development of transonic airfoils.
2. Experimental arrangements

a. Facilities

The experiments have been made in the 150 mm wide blowdown wind tunnel of the Delft University of Technology, Department of Aerospace Engineering. Interchangeable nozzle blocks have been made in that way that a curved transonic flow was obtained in the test section, the radius of the convex lower wall being 500 mm and the height of the test section equal to 90 mm (Fig. 1). By means of a choke far downstream of the test section the shock wave was kept at a position where just in front of the shock wave a Mach number of 1.43 near the lower wall occurred. During the tests the total pressure of the undisturbed flow was 1.89 Bar, the total temperature 270 K. In the test section the static pressure at the lower wall decreased in front of the shock wave more or less linearly in flow direction. Downstream the shock wave the flow endured a subsonic compression over about 100 mm followed by an expansion to sonic conditions at the choke section.

The static pressure field was measured by a static pressure probe. The two pressure taps in this probe were drilled at an angle of 30° with respect to the vertical plane in order to diminish the effects of flow angularity. In fact during all tests the angle of incidence of the probe was less than 3° with respect to the local flow direction.

Because of the normal pressure gradient in the test section a flattened pitot probe was used for the pitot pressure measurements. During these measurements the angle between the probe and the local flow direction was kept less than 5° to avoid substantial errors due to misalignment.

To determine the skin-friction coefficients and the length of the separation bubble a preston probe with outer diameter of 1 mm was used. The wall static pressure to match was obtained from an extrapolation of the static pressures measured by the static pressure probe. All probes were driven by computer controlled stepping motors, making 200 steps for a displacement of 3 mm. The actual position of the probe was determined from the number of steps made from a reference point and taking into account the displacement of the probe due to aerodynamic forces. All pressures have been measured using Druck Ltd pressure transducers.
b. Measurements

The flow in the test section is a transonic flow with a supersonic region upstream of a shock wave and a subsonic region downstream. The position of the shock wave depends on the cross-sectional area of the choke downstream. The effective choke area however will change with the position and the diameter of the probe support protruding through this choke. To check the position of the shock wave, one of the test section side walls was provided with 11 pressure taps in streamwise direction about 70 mm above the lower wall. Ten pressure taps were connected with a scanning valve and were scanned every two seconds, one pressure tap near to the desired shock position was read continuously. As the pressure gradient across the shock wave was very steep, it was possible to adjust the shock wave position by means of the choke within one mm from the target position. To set the choke for this shock wave position, several runs were required because an automatic control was not yet developed at that time.

To obtain the flow quantities vertical traverses were made with both a static pressure probe and a pitot pressure probe upstream and downstream of the shock-wave. Both series of traverses were made in the plane of symmetry at identical streamwise position during different runs of the wind tunnel. Differences in construction of the two probes lead to unequal displacements by the aerodynamic forces on the probes. At the data reduction these differences were taken into account.

To determine the skin-friction a preston probe was used having an outside diameter of 1 mm. It was not possible to make traverses in streamwise direction along the curved lower wall of the test section, therefore the skin-friction measurements were made intermittently. The probe was set at the required position along the surface before the wind tunnel was started and a measurement was made. The proper setting of the choke was checked at each run. Together with the probe pressure measurements the total temperature and the total pressure were measured in the settling chamber.

Visual information about the flow structure was obtained from schlieren and shadow pictures applying continuous light and spark exposures. The spark exposure time was shorter than 50 nanosec (Fig. 2). To check on two-dimensionality of the flow three vertical pitot pressure traverses were made at 70 mm downstream of the shock wave, one traverse in the symmetry plane of the test section and one at both sides at a distance of 30 mm. In the first instance a leakage at one of the side walls caused an a-symmetric flow. After solving this problem, the results of Fig. 3 were measured, still showing some differences in the pitot
pressure distributions. An oil picture of the surface flow just downstream of the shock wave (Fig. 4) indicates the separation line and of the attachment line of the separated flow. At both positions a narrow region of the flow near the plane of symmetry seems to be of a two-dimensional nature. The influence of the wind tunnel side wall boundary layers is revealed by the vortices at both ends of the separation line. Alber, e.a. (Ref. 9) show that a wider two-dimensional region downstream of the shock wave-boundary layer interaction may be obtained by the use of side fences together with boundary layer suction near the side walls. Although these conditions are not fulfilled in the underlying measurements, the results match with earlier investigations (Ref. 2, 3) on shock wave-boundary layer interaction.
3. Data reduction

The results obtained are presented in the rectangular coordinate system of Fig. 5, which is related to the wind tunnel construction. The crest of the convex lower surface was taken as the origin of the coordinate system, the positive x-coordinate being in downstream direction and the positive y-coordinate upwards normal to it. The vertical pitot pressure traverses as well as the vertical static pressure traverses have been made at x-coordinates indicated in this figure. Each probe traverse was started at the lower surface and was ended at about \( y = 40 \text{ mm} \). The Mach number distribution was calculated from the static and pitot pressure distribution. Because the static pressure distributions in y-direction were rather smooth and without large changes, the calculation of the Mach number was made at the data points of the pitot pressure distribution. The static pressures to match were obtained from linear interpolation in the static pressure traverse. From the obtained Mach number distribution all other flow quantities were calculated assuming constant total temperature in the entire flowfield. The boundary layer integral parameters, displacement thickness and momentum thickness, were obtained by numerical integration of the measured flow quantities. In this the edge of the boundary layer was defined as the y-coordinate where the difference between the linear velocity profile of the inviscid outer flow and the measured velocity in the viscous flow region was less than .5%. Although conform to the definition given by Myring (Ref. 10) and Lock (Ref. 11) this procedure is not most frequently found in literature, where the boundary layer thickness usually is related to the position of the highest velocity near the boundary layer edge.

At different stations the skin-friction coefficient has been determined from the law of the wall. In fact at each data point of one vertical traverse the friction-velocity was calculated assuming validity of the law of the wall and at that very part of the traverse where the friction-velocity was found to be reasonable constant the mean value was taken to determine the local skin-friction coefficient. The actual calculations were made with the law of the wall for incompressible flow as suggested by Winter and Gaudet (Ref. 12).

\[
\frac{u_i}{u^*} = 6.05 \log \frac{yu^*}{\nu} + 4.05 \quad (1)
\]

with \( u^* \) the incompressible friction-velocity.
To allow for compressibility effects a correction was applied on the skin-friction coefficient in that way that

\[ c_f^i = \left(1 + 0.2 \frac{M_e^2}{1} \right)^{\frac{1}{2}} c_f \]  

(2)

The boundary layer integral parameter \( \delta^* \) and \( \theta \) have been determined in the usual way, defining the displacement thickness by

\[ \delta^* = \int_0^\delta \left(1 - \frac{\rho u}{\rho e u_e} \right) dy \]  

(3)

and the momentum thickness by

\[ \theta = \int_0^\delta \frac{\rho u}{\rho e u_e} \left(1 - \frac{u}{u_e} \right) dy \]  

(4)

where \( \delta \) is the boundary layer thickness as specified previously in this paragraph.

On behalf of the calculation of the skin-friction with the von Kármán integral relation, Myring (Ref. 10) made two modifications to allow for a normal pressure gradient through the boundary layer. One modification leads to a von Kármán integral relation with an additional pressure term, the second modification resulted in the conventional form of the von Kármán's integral relation, however with modified boundary layer parameters. None of both relations resulted in skin-friction coefficients comparable to those obtained from the law of the wall. Only the latter results have been reported in this paper and some results obtained from preston probe measurements using the Bradshaw-Unsworth formula.
4. Results

a. Flowfield

The measured Mach number distribution, the boundary layer thickness and the location of the separation bubble are presented in Fig. 6. At first sight a close resemblance consists with the results of Seddon (Ref. 2) and Kooi (Ref. 3). At the foot of the shock wave a compression wave from the boundary layer retards the flow to nearly sonic conditions and a second compression wave closes this area. Downstream of this second compression wave part of the flowfield remains supersonic due to a post shock expansion region of which satisfying evidence is given in the lateral pressure distributions of Fig. 9. The two compression waves from the boundary layer coincide at a bifurcation point, meeting the shock wave in the inviscid flow. The curvature of this shock wave increases with increasing distance from the lower surface, which does not correspond with the measurements of Kooi and East (Ref. 3, 5) made at a plane wall. In contrary, they observed a decreasing curvature of the shock wave in a direction away from the bifurcation point.

Due to a difference in shock wave losses above and beneath the bifurcation point, a shear layer was formed in the downstream part of the flow. This shear layer is clearly visible in the schlieren picture of Fig. 7. It starts rather thin behind the bifurcation point and thickens more and more downstream.

The results of the static pressure traverses have been given in Fig. 8 in the cartesian coordinate system of Fig. 5. Outside of the direct shock wave region the normal static pressure gradient is rather constant, inside as well as outside the boundary layer. Near the shock wave its influence is felt and at the station \( x = -15 \) mm it was even not possible to get reliable information about the static pressure above \( y = 15 \) mm.

The static pressure distributions in streamwise direction at different distances from the lower wall given in Fig. 9 were obtained from the vertical static pressure traverses. With increasing distances from the surface the pressure jump across the shock wave increased just as the strength of the post shock expansion. The wall pressure distribution (Fig. 9) is obtained from an extrapolation of the vertical static pressure traverses, no pressure holes in the surface were available at that time. Noteworthy is the pressure fall just in
front of the shock wave, the relative small pressure rise across the shock wave and the lack of evidence about a post shock expansion at this level. Downstream of the shock wave the static pressure rises gradually to a more or less constant value. The pitot pressure distributions measured at identical stations as the static pressure traverses are given in Fig. 10. The measurements upstream of the shock wave indicate an onset to separation at \( x = -20 \text{ mm} \) and a clear evidence for separation at \( x = -15 \text{ mm} \). At the latter station the drop in pitot pressure across the shear layer due to the shock bifurcation is indicated by the symbol "\( \omega \)". The first pitot pressure distributions downstream of the shock wave start normal to the \( x \)-axis with a value being constant for several data points. The separation started at \( x = -15 \text{ mm} \) continues up to \( x = 30 \text{ mm} \). The pitot pressure distributions further downstream give no evidence of separation, so at \( x = 40 \text{ mm} \) the flow may be considered to be attached again. The location of the shear layer in this part of the flowfield is also indicated by the symbol "\( \omega \)".

From the static pressure and the pitot pressure traverses the velocity distributions have been calculated assuming constant total temperature through the whole flowfield. Fig. 11 gives the obtained velocity profiles. The boundary layer thickness as defined before is indicated by "\( \delta \)". Near the shock wave just outside the boundary layer there is a staggering of data points, which may be an indication for the unsteady behaviour of the probes in this part of the flowfield. Also the determination of the boundary layer thickness in this part of the flowfield is open for arguments. It is rather difficult to determine the undisturbed velocity distribution outside the boundary layer, no more than to decide which part of the flowfield belongs to the boundary layer, so the indicated boundary layer thickness is more or less arbitrary. The velocity distributions downstream of the shock wave seems to be more reliable, because the data points are closer to a smooth velocity distribution. Near the wall surface the velocity is nearly zero for several data points, with exception of the stations \( x = -10 \text{ mm} \) and \( x = 0 \text{ mm} \). The boundary layer is undoubtedly separated, but the precise length of the separation is difficult to determine from these velocity profiles having rather big distances between the stations, the skin-friction distribution will give better evidence.

Since the height of the separation bubble (Fig. 6) changes only gradually in streamwise direction, the measured height of 1.3 mm for the zero velocity streamline will be a proper result for half the bubble height. This means a separation bubble height of 2.6 mm, which is about .4 times the undisturbed boundary layer thickness. This result is smaller than found in the investigations of Seddon (Ref. 2) and Kooi (Ref. 3). The last four velocity profiles of Fig. 11
are also given as non-dimensionalized profiles in Fig. 12. Although these velocity profiles may not be considered to be fully developed turbulent boundary layer profiles, they still are partially logarithmic.

b. Boundary layer integral parameters

The development of the boundary layer thickness $\delta$ in the interaction zone is given in Fig. 13. Upstream of the shock wave $\delta$ decreases a little in streamwise direction, but downstream of the shock the boundary layer grows up to about 1.75 times its original value. Further downstream a gradual growth up to 2.25 $\delta_o$ has been measured and as far as may be concluded from the present investigation this boundary layer growth will endure. The local reduction in boundary layer thickness downstream of the shock wave at $x = 20$ mm is conform to the results of similar investigations made at plane wall interactions (Ref. 2, 3, 6). In Ref. 3 this reduction only has been found at a lower Mach number of 1.40, where the separated region was relatively short. At a higher Mach number no reduction in boundary layer thickness was found in spite of the more extended separated region. In Ref. 6 a reduction in boundary layer thickness was measured only at a low Reynolds number ($Re_{shock} = 9 \times 10^6$), at a Reynolds number of $36 \times 10^6$ no reduction of the boundary layer could be detected.

The displacement thickness (Fig. 14) shows a rapid growth near the shock wave. Downstream of the shock a maximum value of 9.5 times the undisturbed displacement thickness has been reached at $x = 40$ mm, after which a gradual reduction starts.

The overhaul feature corresponds to results obtained from measurements at a plane wall, except that part of the interaction ($x = 10$ until $x = 40$) where the growth in displacement thickness reduces remarkably. This might be accounted to the normal pressure gradient which suppresses the development to a usual type of separated boundary layer. The maximum value of the displacement thickness $\left(\frac{\delta^*}{\delta_o} = 8.5\right)$ is higher than the values obtained from shock wave boundary layer interactions at a plane wall at similar conditions. In Ref. 2 a maximum displacement thickness of $\left(\frac{\delta^*}{\delta_o} = 5.5\right)$ was found.

The momentum thickness (Fig. 15) in the interaction region increases gradually in downstream direction and agrees in nature with the results from plane wall experiments. The present values however are twice as high and might even be higher in the separated region if the contribution of the separation is well taken into account. In that case a continuous growth up to a value of nine times

the undisturbed momentum thickness may be expected at $x = 80$. This value is nearly twice as high as found in the experiments of Kooi (Ref. 3).

The shape factor, calculated from the measured displacement thickness and the measured momentum thickness is given in Fig. 16. Due to irregularities in the value of the momentum thickness between $x = -25$ and $x = 10$ the obtained shape factor is rather erratic in this part of the flowfield. To get more reliable results an extension of measurements in this part of the flow is needed. Upto the position where separation starts the shape factor changes smoothly. In literature it is commonly accepted that the flow will separate at a shape factor of 2.6 (Seddon, Alber). In the present measurements this value is reached at station $x = -20$, at which separation is indeed detected by oil flow and by the shear stress distribution obtained from the law of the wall. The maximum value of the shape factor $H = 5$ is reached at $x = 5$, which coincides with the position half way the separation bubble. After this the shape factor seems to decrease to a constant value of $H = 2$ at $x = 80$. Reattachment of the flow occurred in between at a position where the shape factor was decreased to a value of 3.5.

c. Skin-friction measurements

Two methods have been used to determine the skin-friction coefficients from measured quantities.

The first one is based on the assumption of a logarithmic part in the velocity distribution of the turbulent boundary layer. At every data point the friction-velocity was calculated from Eq. 1 using the local values of $u$, $v$ and $v_e$. The mean value of a region where a constant value for the friction-velocity was found has been used to calculate the local skin-friction by means of Eq. 3. The results of these calculations are given in Fig. 17. The calculated values of the skin-friction at $x = -20$ and $x = -25$ are rather arbitrary. At both stations the results depend mainly on that data point which was nearest to the wall. The reliability of these points is rather doubtful. With this in mind the skin-friction distribution along the wall has been drawn including the separated region near the shock wave. Upstream of the separation the skin-friction coefficient decreases over a short distance to a zero value. Downstream of the separation the skin-friction coefficient increases gradually, but at $x = 80$ its value is still only one third of the undisturbed value upstream of the interaction.
The second method applied for skin-friction determination was by preston probe measurements. The matching static pressures were obtained from extrapolations of the static pressure probe readings. With the static pressure and preston probe readings the skin-friction was determined using the Bradshaw-Unsworth formula (Ref. 13). These results also have been plotted in Fig. 17. Upstream of the separation these skin-friction coefficients are higher than the law of the wall results, but downstream they do not fit at all with the exception of $x = 30$.

Unfortunately, the preston probe measurements have been stopped between $x = 20$ and $x = 30$ without giving an indication about reattachment of the flow. Still the conclusion may be drawn that the length of the separation bubble obtained from preston probe measurements is shorter than that obtained from the law of the wall. Oil flow technique sustains the last results just a schlierens-pictures do, but which of them are the most reliable is still uncertain. A rough estimation of the length of the separation results in about 45 mm, giving a dimensionless separation length $\frac{\delta_s}{\delta_0} = 7.4$. Compared to data of various investigators given in Fig. 18 the separation length is shorter than found from plane wall interactions at similar Mach number with exception of Seddon's experiment, in case a correction for Reynolds number is allowed. The decrease of separation length will certainly not be generated by the positive pressure gradient along the wall. Such a pressure gradient will rather increase the length of the separation. Therefore the normal pressure gradient might be responsible for shortening the length of the separation and dominate the influence of the adverse pressure gradient. In his study about the influence of wall curvature Inger (Ref. 7) finds a little higher skin-friction coefficient at its lowest value in the flowfield, which may indicate a small delay of the onset to-incipient separation and probably an indication for a shorter separation length.
5. Conclusions

The interaction between a turbulent boundary layer at a convex wall and a normal shock wave have been investigated in a curved transonic flow at a Mach number of 1.43.

Conclusions drawn from this investigation are:

- At a Mach number of 1.43 just in front of the shock wave, a wall curvature \( \frac{R}{\delta_0} = 80 \) and a Reynolds number \( \text{Re}_\delta = 2 \times 10^5 \), separation occurs at a shape factor \( H = 2.6 \).

- The length of the separated region \( \left( \frac{y_s}{\delta_0} = 7.4 \right) \) is smaller than usually found at a plane wall. The influence of the adverse pressure gradient on the length of the separated region is obviously overruled by the normal pressure gradient affecting the eddy viscosity of the turbulent boundary layer.

- The height of the separation bubble \( (.4 \delta_0) \) is smaller than found by Seddon and Kooi at a plane wall.
Notation

c_f  skin-friction coefficient
H    shape factor
K    degrees Kelvin
l    separation length
M    Mach number
p    pressure
Re   Reynolds number
u    velocity
u_T  friction velocity
x, y, z coordinates
w    wake

Indices

e    edge
i    incompressible
o    initial undisturbed value
s    separation

Greek symbols

δ    boundary layer thickness
δ*   displacement thickness
ρ    density
θ    momentum thickness
v    viscosity
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FIG. 1: NOZZLE BLOCKS.

FIG. 2: SPARK SHADOGRAPH OF THE FLOWFIELD.
FIG. 3: THREE PITOT PRESSURE DISTRIBUTIONS DOWNSTREAM OF REATTACHMENT
FIG. 4: OIL FLOW PICTURE OF SEPARATED REGION

FIG. 5: COORDINATE SYSTEM AND LOCATIONS OF PROBE TRAVERSES
FIG. 7: SCHLIEREN PICTURE OF THE FLOW FIELD
FIG. 8: STATIC PRESSURE MEASUREMENTS
FIG. 9: STATIC PRESSURE DISTRIBUTIONS AT DIFFERENT DISTANCES FROM LOWER WALL
FIG. 11: VELOCITY PROFILES
FIG. 12: NON-DIMENSIONALIZED VELOCITY PROFILES AT x = 50, 60, 70, 80 mm.
FIG. 13: BOUNDARY LAYER THICKNESS

FIG. 14: BOUNDARY LAYER DISPLACEMENT THICKNESS.
FIG. 15: MOMENTUM THICKNESS.

$\theta_0 = 0.195 \text{ mm}$

FIG. 16: SHAPE FACTOR H.
Fig. 17: SKIN FRICTION DISTRIBUTION
**FIG. 18: EFFECT OF REYNOLDS NUMBER ON THE SEPARATION LENGTH.**