An Investigation of a Strong Shock-Wave Turbulent Boundary Layer Interaction in a Supersonic Compressor Cascade

H. A. SCHREIBER and H. STARKEN
Deutsche Forschungsanstalt fur Luft- und Raumfahrt e.V.
Institut fur Antriebstechnik
Linder Hohe, 5000 KölN 90, Germany

ABSTRACT

Experiments have been performed in a Supersonic cascade facility to elucidate the fluid dynamic phenomena and loss mechanism of a strong shock-wave turbulent boundary layer interaction in a compressor cascade. The cascade geometry is typical for a transonic fan tip section that operates with a relative inlet Mach number of 1.5, a flow turning of about 3 degrees, and a static pressure ratio of 2.15. The strong oblique and partly normal blade passage shock-wave with a pre-shock Mach number level of 1.42 to 1.52 induces a turbulent boundary layer separation on the blade suction surface. Freestream Reynolds number based on chord length was about 2.7x10^6. Cascade overall performance, blade surface pressure distributions, Schlieren photographs, and surface visualizations are presented. Detailed Mach number and flow direction profiles of the interaction region (lambda shock) and the corresponding boundary layer have been determined using a Laser-2-Focus anemometer. The obtained results indicated that the axial blade passage stream sheet contraction (axial velocity density ratio) has a significant influence on the mechanism of strong interaction and the resulting total pressure losses.

NOMENCLATURE

\( AVDR \) - axial velocity density ratio
\( AVDR = \frac{\rho_v w_2 \sin \beta_2}{(\rho_v w_1 \sin \beta_1)} \)

\( c \) - chord length
\( H_{12} \) - boundary layer shape factor
\( H_{32} \) - boundary layer shape factor
\( M \) - Mach number
\( M_{1s} \) - isentropic Mach number
\( P \) - static pressure
\( P_{2}/P_1 \) - static pressure ratio across cascade
\( \rho \) - density

\( R \) - radius of curvature on blade suction surface

\( Re \) - Reynolds number, \( \frac{w_1 c}{\nu} \)
\( t \) - blade spacing
\( w \) - flow velocity
\( x \) - coordinate along blade chord
\( y \) - (a) coordinate normal to chord
(b) distance to the blade surface and normal to \( \beta_{\text{Ref}} = 150^\circ \)

\( \beta \) - flow direction with respect to cascade front
\( \Delta \beta \) - flow direction deviating from \( \beta_{\text{Ref}} = 150^\circ \)
\( \beta_s \) - boundary layer thickness
\( \delta \) - displacement thickness
\( \delta_1 \) - momentum thickness
\( \delta_2 \) - energy loss thickness
\( \eta \) - coordinate in tangential direction
\( \nu \) - kinematic viscosity
\( \omega \) - total pressure loss coefficient
\( \rho \) - density

INDICES

0 - condition just upstream of interaction region
1 - uniform inlet conditions
2 - uniform exit conditions (mixed out)
\( \text{is} \) - isentropic
\( \delta \) - condition at outer edge of boundary layer

INTRODUCTION

The trend towards higher pressure ratios and compact aircraft engines with a reduced number of stages leads to a considerable increase of aerodynamic loading of the fan and compressor stages. Thereby the velocities relative to the blades increase to transonic and supersonic speeds and shock-waves occur in front and within the blade passages. The aerodynamic...
The performance of such fan and compressor blades is essentially dependent on both the total pressure losses of the shock-waves and the momentum losses and displacement effects of the boundary layers on the blade surfaces. Therefore, the interaction of the shock-waves with the boundary layer becomes an aerodynamic problem of immense practical importance. The shock-waves with their strong adverse pressure gradient cause considerable modification of the boundary layer itself, as well as of the flow field external to the boundary layer. For blade-to-blade aerodynamics the latter is of paramount importance, since due to the interaction, additional blockage can change the blade passage throat area and may even cause unexpected choking. The starting phenomena of supersonic blades, for example, are essentially dependent on the shock-wave/boundary layer interaction mechanism.

Blades of highly loaded fans or core compressors often operate with high relative inlet Mach numbers whereby boundary layer separation in combination with shock interaction cannot be avoided. The designer of such blades, however, would like to know how the cascade geometry and, in more detail, the loading distribution and shock-wave pattern have to look in order to meet the design goal and to avoid unacceptable loss increases.

Recent progress in the theoretical blade-to-blade methods (inviscid-viscous interaction as well as more sophisticated Navier Stokes codes) are very encouraging, but still the transonic flow fields with strong imbedded shock-waves and boundary layer separations create tremendous difficulties. More detailed experimental data of such highly loaded configurations can help to improve the necessary theoretical models.

To elucidate the fluid dynamic phenomena and loss mechanism of the transonic blade-to-blade flow, and especially the structure of the interaction of a strong passage shock-wave with the blade boundary layer, an experimental investigation on a compressor cascade was performed in a supersonic cascade facility. The investigations performed on the blade suction surface were to a certain degree similar to the detailed experiments on strong shock-wave/boundary layer interactions of Seddon (1960), East (1976), Mateer (1976), Kooi (1978), Delery (1978), and Xiu and Squire (1987). However, the experimental setup was installed in such a way that a real compressor blade element flow could be simulated. Thereby the boundary layer development upstream, across, and behind the strong shock-wave of this model corresponds to the boundary layer behavior in a realistic blade configuration.

This is especially important since the interaction mechanism with boundary layer separation is strongly influenced by the flow around the free trailing edge. Basic shock-wave/boundary layer interaction experiments on tunnel walls or on flat plates would not be able to provide data relevant to turbomachinery blade-to-blade flows.

CASCADE BLADE DESIGN

A typical fan blade tip section with low camber and a relative blade thickness of 3.5 percent was designed for this investigation. At an inlet Mach number of 1.5 the cascade has to decelerate the flow to subsonic velocities providing a static pressure ratio of 2.15 and a flow turning of 3 degrees. In order to reduce the Mach number incident to the first passage shock-wave, the blade was designed with a negative suction surface camber along the cascade entrance portion. Thereby the velocity is reduced isentropically to a Mach number level around 1.4 over most of the covered passage entrance. This so-called pre-compression design allows a considerable reduction of shock losses resulting from the detached bow-shock and the first shock at the blade passage entrance. The first passage shock was designed to be oblique with a deflection of 8.4 degrees and a deceleration to nearly sonic velocities.

The waves emanating from the front portion of the blades propagate upstream axially and establish the so-called unique incidence condition. Thereby the supersonic flow into the blade passage is started and the back pressure can be varied without influencing the upstream flow condition.

Fig. 2 Schok-wave pattern at design condition

Fig. 3 Openings for sidewall suction
Fig. 2 shows a sketch of the shock-wave pattern that leads to an overall static pressure ratio around 2.15. At this back pressure condition the first passage shock is partly normal and causes a strong boundary layer separation forming a typical lambda shock system above the separated region. The flow behind the first oblique passage shock reaccelerates slightly along the front portion of the blade pressure surface from Mach 1.05 to Mach 1.30, and is decelerated finally to subsonic velocities by a second nearly normal passage shock.

TEST FACILITY AND INSTRUMENTATION

Cascade Facility and Test Condition
The tests were performed in the DLR supersonic cascade wind tunnel at Köln-Porz. This tunnel is a closed-loop, continuous running facility, equipped with an adjustable converging-diverging nozzle allowing Mach number variations from 1.3 to 2.4. For the tests the wind tunnel was operated with a total pressure in the settling chamber around 1.2 bar and a total temperature of 319 K. The freestream Reynolds number based on chord length was around \(2.7 \times 10^6\) and the appropriate average Reynolds numbers at the beginning of the strong interaction region (\(x = 100\) mm from blade leading edge) were \(Re_{\infty} = 1.66 \times 10^7\) and \(Re_{\delta} = 0.3 \times 10^7\). Some aerodynamic test data are summarized in Tab.1 and Tab.2 provides some estimated uncertainties for key dependent variables and laser data. The boundary layer ahead of the interaction region is assumed to be turbulent after natural transition on the front portion of the blade. To improve the measurement accessibility to the shock-wave/boundary layer interaction region and to the boundary layer, the blade chord was selected to be 170 mm. Due to the limited size of the tunnel, only three blades with 152 mm span could be installed in the test section.

Most of the former cascade tests and many of the shock-wave/boundary layer interaction experiments suffered from detrimental wall effects. The flow field, at least behind the strong shock-wave, is strongly influenced by thickening of the sidewall boundary layers, so the development of pure two-dimensional, or at least quasi two-dimensional flow conditions is prevented. To overcome this problem, at least partly, a sidewall boundary layer suction was applied in the area where the first passage shock interacts with the wall boundary layer. The suction allowed a controlled boundary layer removal through trapezoidal openings which had been cut in the plexiglas windows (Fig. 3). In the exit area of the cascade the flow is guided by two tailboards which are hinged to the trailing edges of the upper and lowermost blades. With throttles located at the downstream end of the tailboards, the desired back pressure and corresponding shock-wave pattern could be adjusted. Using this tailboard/throttle system, a very stable shock-wave position could be achieved. Shadowgraphs, Schlieren observation, and the blade surface pressure distribution were used to monitor the exact shock-wave position. Periodicity in the inlet flow region of supersonic cascades is not an essential problem because the periodic flow pattern is automatically established behind the first blade, as long as the cascade operates with supersonic started flow and the nozzle flow is uniform. The outlet flow periodicity was checked by sidewall static pressures and probe traverses behind the center blade and was found to be reasonably good.

Instrumentation
To determine the overall performance of the cascade, wake traverse measurements were performed at midspan location using a combination probe for static pressure, total pressure, and flow direction. The probe was located at an axial distance of 15 mm (x/c = 0.167) downstream of the cascade exit plane. The two neighbouring blades (2nd and 3rd) which form the investigated blade passage were instrumented to measure the blade surface pressure distribution.

A non-automated Laser-2-Focus anemometer (L2F) (Schoedl, 1980) was used for velocity and flow angle measurements in the region of shock-wave/boundary layer interaction, which included the blade suction surface boundary layer at different midspan positions downstream of 58 percent of chord. The L2F anemometer operated in a back scatter mode and had the following characteristics:

- Argon-Ion laser with a power of about 300 mW
- Focal length of the receiver optics 343 mm
- Lens diameter of the receiver optics 100 mm
- Diameter of focused beams 12 \(\mu m\)
- Separation of beams \(s = 332.6 \mu m\)

To improve the intensity of the focused laser beams, especially for measurement stations near the blade surface, the axis of the optical arrangement was inclined by 3 degrees with respect to the spanwise direction. Thereby a cut-out of the illuminating laser beams by the blade surface could be avoided. Oil mist was produced by a droplet generator and introduced into the settling chamber to improve the signal rate in the measurement volume. Due to their small size the oil droplets (about 0.07 \(\mu m\) in mean diameter) adequately followed the flow, including the strong decelerations across shock-waves. The L2F system allowed measurements in the boundary layer as near as 0.1-0.2 mm to the blade surface (in the undisturbed nonseparated boundary layer). With increasing turbulence intensity in the boundary layer, however, the measuring time increased considerably, and beyond a certain turbulence level (about 20-30 percent) the useful signals could not be distinguished from the overall signal noise. As the turbulence level increases considerably in separating and especially at the edge of the recirculation region, no signals were obtained below a certain distance from the blade surface for measurement stations downstream of the shock (see Fig.14). In addition, it should be mentioned here that the turbulent shear layer has a dispersive effect on the seeding particles, which are transported in the boundary layer so that the measurement rate is much lower near the wall.

To overcome the disadvantage of the L2F system in the high turbulence region and close to
Table 1  Summary of Aerodynamic Test Data

Cascade data:

- Total pressure \( p_1 = 1.2 \text{ bar} \)
- Total temperature \( T_{t1} = 319 \text{ K} \)
- Upstream Mach number \( M_1 = 1.5 \)
- Upstream flow angle \( \beta_1 = 150^\circ \)
- Chord Reynolds number \( Re = 2.7 \times 10^6 \)

Boundary layer and shock interaction data on blade suction side:

- B. 1. transition at \( x/c = 0.17 \)
- Shock-wave position (normal part) \( x_s/c = 0.65 \)
- Surface curvature in the interaction region \( R_{ss}/\delta = 85 \)
- Pre-shock Mach number \( M_0 = 1.44-1.52 \)
- Undisturbed boundary layer (x=98 mm) \( \delta_0 = 2 \text{ mm} \)
- Reynolds number \( Re_0 = 0.3 \times 10^5 \)
- B. 1. separation at \( x/c = 0.64-0.65 \)
- Increase of displacement thickness in the interaction region \( \delta_1/\delta_0 = 6.5 \pm 10 \)

Table 2  Estimated uncertainties

- Upstream Flow angle, \( \beta_1 \) ± 0.3 deg
- Exit flow angle, \( \beta_2 \) ± 1.0 deg
- Loss coefficient, \( \omega \) ± 0.005
- AVDR ± 0.03
- Laser data: velocity, \( w \) ± 0.5-1.5%
- flow direction ± 0.3 deg

TEST PROCEDURE

After the flow and desired upstream Mach number were stabilised in the closed loop tunnel, the back pressure was adjusted by using the tabstock tailboard throttle system and by regulating the amount of sidewall boundary layer suction through the openings in the sidewalls. The axial stream sheet contraction at midspan, expressed by the axial velocity density ratio (AVDR), results from the endwall boundary layers thickening within the blade passages, and can be controlled partly by the amount of sidewall suction. Blade pressure distributions and wake traverse data were recorded simultaneously with the L2F anemometer and Pitot probe measurements in the boundary layer. This simultaneous data acquisition was important since due to the transonic velocities within the cascade blade passage, the flow field was extremely sensitive to any change in the boundary conditions (throttle setting, endwall suction).

RESULTS

Description of the Strong Interaction Region

The investigations were performed at a cascade inlet Mach number of about 1.5 and a constant backpressure. The shock-wave pattern of the precompression-blade cascade is sketched in Fig.2, and a Schlieren photograph representative of the tests done while making boundary layer measurements is provided in Fig.4.

In front of the leading edge a detached bow shock develops, that has a weak oblique extension into the upstream region and a stronger oblique extension, with about 8.5-9 degrees of flow deflection, running into the covered passage. Due to the prescribed back pressure a quasi-normal shock-wave is formed near the suction surface, where it interacts with the boundary layer. Boundary layer separation accompanied by a lambda shock system, with its leading oblique shock and rear quasi normal shock (see Figs.4 and 5), characterizes the interaction region. Basically this lambda shock system and the corresponding boundary layer behaviour is very similar to the findings in
The classical experiments of Seddon (1960), Kool (1978), and East (1976). The essential difference is, however, that this shock system is established in a real blade passage and that the lambda shock develops above a relatively strong convex curved part of the blade surface. The radius of curvature within this region is of the order of the blade chord (R/c=1.0), and relative to the undisturbed boundary layer thickness ahead of the interaction region this radius is about R/δ₀ = 85.

The quasi normal shock has two bifurcation points; the lower one belongs to the lambda system and the upper one forms as part of the Mach reflection of the oblique passage shock (discussed below). Due to different velocities and densities behind the shocks, vortex sheets develop downstream of these bifurcation points fading away rather slowly in the streamwise direction. The formation of the quasi-normal shock-wave in the flow field and its bifurcation points can also be interpreted as a Mach reflection phenomenon. It occurs when the intensity of the oblique passage shock increases to such a level that a regular reflection of the oblique passage shock is not possible. Beyond a certain intensity or deflection angle near the suction surface this oblique passage shock splits into a quasi normal shock that interacts with the suction surface boundary layer (forming the lambda shock system), and into a rear reflected oblique shock that starts at the upper shock bifurcation point. The corresponding strong flow deflection that is necessary for the Mach reflection near the blade surface in turn is caused by the wedge-type displacement effect of the severe boundary layer separation.

The Mach numbers ahead of the oblique passage shock are nearly constant with a level around 1.4, but due to the convex curvature of the suction surface just upstream of the lambda shock (x/c>0.5), the pre-shock Mach numbers increase towards the blade surface and reach a maximum value of about 1.52 at the edge of the boundary layer (deduced from the blade surface pressures). Schematic drawings of the flow field interpretations of a test series are given in Figs.5 and 6. The shock locations, flow vectors, and data were derived from laser anemometer surveys. The figures show the derived shock structure, the edge of the boundary layer, the variation of the displacement thickness, and the sonic line (M=1.0) within the boundary layer. At some characteristic positions, measured Mach numbers and flow angles are indicated (Fig.5). Across the leading oblique shock-wave of the lambda system, for example, the flow is turned away from the blade surface by about 8.5 degrees and decelerated from Mach number 1.5 to about 1.19. The rear leg of the lambda shock seems to be a slightly curved, strong oblique shock, decelerating the flow to high subsonic Mach numbers (M=0.85-1.0).

The quasi-normal shock above the lower bifurcation point reduces the Mach numbers to about roughly 0.75-0.77. It should be mentioned here that all Mach number and flow angle data indicated in Fig.5 are data directly obtained from the L2F anemometer and are not corrected by any use of theoretical shock relations. These data therefore include the inaccuracies inherent in the measurement system, including the effect of the seeding particles not following the strong decelerations across the shock-waves completely, although the particle lag is extremely small. Just downstream of the shock system there is a relatively strong reacceleration to sonic and slightly supersonic velocities, which is an effect also observed in a previous independent test series (Schreiber, 1987) in which streamwise Mach number and flow angle variations across the shock system were also obtained using an L2F anemometer. One example from those results is provided in Fig.7, which shows clearly the Mach number and flow angle variations across the lambda shock system slightly above the viscous layer (y=5 mm). The post-shock expansion behind the rear leg of the lambda system happens within a distance of only 4 mm, or 2°.
boundary layer and everywhere behind the rear Mach number of 1.47 a region of locally super-sonic throughout the strong interaction region, as indicated in Fig.5. This phenomenon raises an interesting question, namely, whether there exists a so-called "supersonic tongue" downstream of the rear leg of the lambda shock system, an effect observed in the results of Seddon (1960). He found that for a pre-shock Mach number of 1.47 a region of locally supersonic flow develops near the edge of the boundary layer and everywhere behind the rear leg of the lambda shock system with a downstream extension of several boundary layer thicknesses. The results obtained from the present laser anemometer analysis show a similar phenomenon along the boundary layer edge.

However, contrary to Seddon's results, most of the flow downstream of the rear shockleg is subsonic (see Mach number profile at x = 115.5 mm in Fig.14). The present results, however, are in good agreement with the findings of East (1976) and Kool (1978), who recognized that this local supersonic region develops for pre-shock Mach numbers beyond 1.4-1.44 and is concentrated only on the vicinity of the boundary layer and the lowest branch of the rear shock.

The structure and the downstream extension of the supersonic region behind the rear leg strongly depends on the conditions prescribed by the boundaries of the rear flow field, i.e., by the geometry of the rear flow channel and the prescribed back pressure. In the former flat plate experiment the "supersonic tongue" closes after a certain downstream distance because the overall flow is decelerated to subsonic flow. In the present cascade experiment the small supersonic region spreads out due to a slight overall reacceleration in the blade passage. Only local subsonic flow regions exist behind the rear leg of the lambda shock and behind the quasi-normal shock-waves above the two shear layers, as indicated in Fig.6. Regarding the loss mechanism of this cascade passage flow, it is interesting to note that behind the region of this strong bifurcated shock system (about 25-30% of blade channel height) the maximum level of shock losses can be found (see Fig.6).

Besides the back pressure, which strongly influences the rear blade passage flow field, the most important parameter is the pre-shock Mach number; its level determines the growth and the downstream extension of the separation region and its displacement. The separated boundary layer forms a so-called "ramp", the magnitude of which determines the strength and extension of the leading oblique shock and the corresponding height of the lower shock bifurcation point.

Incresing the pre-shock Mach numbers above 1.4, therefore, cause an enlargement of the whole lambda shock system.

Blade Pressure Distribution

The blade surface Mach number distribution, deduced from surface static pressures and platinum total pressure, is shown in Fig.8. Typical for this type of blade section, i.e., with a concave front portion, it is the strong deceleration just behind the leading edge, thereby reducing the suction surface Mach number from a peak value of about 1.67 to a level around 1.4. Within this region boundary layer transition occurs at about 17% of chord. (The Mach number distribution between about 25 to 45 percent chord does not exactly correspond to the design distribution, as it is slightly influenced by some disturbances emanating from the tunnel sidewalls.) A convex curvature starting at the midportion of the suction surface (49 to 60 percent chord) reaccelerates the flow to a Mach number of 1.52 incident on the lambda shock.

At the onset of interaction there is a steep rise in pressure within a distance of 3-4 percent of chord (5-6 mm). Downstream of this sharp initial increase follows a region of more gradual pressure rise remaining nearly linear until the trailing edge is reached. At transition between steep and moderate pressure gradient ("kink" point) the boundary layer separation point can be assumed. This point is located about half way between the leading oblique shock and the rear leg of the lambda shock system. As visualized by oil and ink traces (see Fig.13 top) the boundary layer seems to be fully separated throughout the rear portion of the lambda leg is of interest to note that in spite of the severe separation, the suction pressure distribution shows a considerable flow deceleration in the rear part of the covered blade passage.

When comparing in Fig.10 the blade surface Mach numbers, deduced from the surface pressures with the Mach numbers at the outer edge of the boundary layer, obtained from the L2F anemometer, the L2F data show considerably lower values than the blade surface data.
However, it must be considered that the blade pressure data are represented as "isentropic" Mach numbers since the actual local total pressure is not known, whereas the Mach numbers at the edge of the boundary layer are actual local Mach numbers deduced from the measured flow velocities. The "isentropic" blade surface Mach numbers can be corrected approximately to "actual" Mach numbers by assuming a total pressure loss that roughly corresponds to a strong oblique shock ($p_2/p_1 = 0.96$, see Fig.10). It is also interesting to note that the L2F data at the boundary layer edge indicate a stronger deceleration across the lambda shock system, followed by a slight post shock expansion.

**AVDR Influence**

When the supersonic flow into the cascade blade passage is started, the axial velocity density ratio (AVDR) can be varied in a test series while keeping the static pressure ratio constant. Investigations of this sort with the present cascade showed that the AVDR may play an important role in shock boundary layer interaction and the corresponding loss generation. Tests involving L2F measurements in the interaction region were performed with AVDR values of 1.07 and 1.13, respectively. The results showed that the lower AVDR condition produced a stronger boundary layer separation, a higher shock bifurcation point of the lambda shock system, and slightly higher overall losses ($\omega = 0.14$ compared to $\omega = 0.12$ for the test with higher AVDR). Increased stream tube convergence (higher AVDR) shifted the shock system within the blade passage slightly upstream, as illustrated in Fig.11, and the corresponding near-wake total pressure data in Fig.12 show lower losses downstream of the blade suction side. Also, the blade Mach number distributions provided in Figs.8 and 9 can be compared, revealing the differences in the location of the second passage shock, as well as slightly different pressure gradients on the suction surface behind the lambda shock. Stronger separations at lower AVDR conditions.
appear to reduce the static pressure recovery along the rear part of the suction side.

The observed AVDR influence on shock-wave position and losses is in good general agreement with former observations in compressor cascade tests, where an increased AVDR, that is a contraction of the blade passage stream channel, causes a reduction of supersonic Mach numbers, a slight upstream shifting of the passage shocks (even when the back pressure is forced to be constant), and a relief of the diffusing subsonic flow in the rear of the blade passage. Therefore, an increase of the AVDR results in a slight reduction of overall total pressure losses, the reduction due to decreased viscous losses in the rear passage, as well as to decreased shock losses.

Surface Flow Visualisation

In some initial tests, surface streak techniques were used to study the boundary layer behaviour. Fig.13 shows an oil flow pattern on each blade surface and a corresponding Schlieren photograph for a test with an upstream Mach number around 1.49 and a static pressure ratio around 2.0. In the front portion of the blade where the suction surface flow (Fig.13, top) is decelerated along the concave blade contour, a region of boundary layer transition from laminar to turbulent flow is indicated. Transition is complete around 16-17 percent blade chord. Downstream of about 63-67 percent of chord a separated flow region is indicated. However, it is very difficult to determine a well defined separation line or a line of boundary layer reattachment from these oil streak patterns. The direction of reverse flow can be clearly seen, starting at the trailing edge. Also, some vortices can be seen in the left and right corner regions, indicating that there is some three-dimensionality within the separated region. This three-dimensionality is partly induced by the sidewall boundary layer suction starting downstream of 57 percent chord. The oil streak pattern on the blade pressure surface (Fig.13, bottom) shows oil accumulating...
in the region where the rear passage shock interacts with the blade boundary layer at about 38 percent chord.

A second technique more appropriate for studying flow separation is the ink injection method. Thereby tiny streams of ink are allowed to enter through inclined drilled holes on the blade surface. At flow separation regions the ink spreads in the spanwise direction and forms a well-developed line. The technique also indicates the direction of reverse flow. Ink injection tests confirmed that no reattachment happened downstream of the lambda shock system.

Velocity Profiles and Integral Properties

Mach number profiles for the strong interaction region are provided in Fig.14. The data were obtained using the L2F anemometer and for an AVDR of 1.07. At each measurement point the local flow direction is indicated. Note the shape and development of the Mach number profile close behind the rear leg of the lambda shock (x=115.5 mm and 120.5 mm). The small supersonic flow region at the edge of the boundary layer ("supersonic tongue") spreads out in a relatively short distance of 5 mm (about 2.5 δs) during which the flow is turned simultaneously by about 7 degrees. Furthermore, the positions of the leading oblique shock of the lambda system and the shear layer downstream of the shock bifurcation point are readily apparent.

As mentioned earlier, the laser system encountered difficulties in recording data in regions of very high turbulence and close to the wall, i.e., where boundary layer separation occurred. Therefore, in these regions the shape of the boundary layer was not measured. Nevertheless, integral boundary layer parameters have been determined from the measured profiles in order to show the boundary layer behaviour on the rear part of the blade suction surface.

Figs.15 and 16 provide the integral parameters at the two AVDR conditions, 1.07 and 1.13, respectively. Upstream of the interaction region (x=98 mm) the boundary layer has a thickness of about 2 mm and a shape factor H12 of around 2.4. As the lambda shock system is...
approached a rapid increase of the displacement thickness $\delta$ is observed. This increase starts when the flow at the edge of the boundary layer passes the leading oblique shock and is turned away from the blade surface by about 8-9 degrees. About 9-10 boundary layer thicknesses downstream of the start of interaction the shape factor $H_{12}$ reaches a maximum value, and the displacement thickness is increased by a factor of about 6.5 at an AVDR of 1.13, and a factor of 9 at an AVDR of 1.07.

Beyond the maximum value of $H_{12}$ the boundary layer rehabilitation phase starts. In contrast to the classical flat plate experiments, the boundary layer in the present tests had to withstand a continuous pressure increase down-stream of the shock system, resulting in a further, nearly linear increase of the displacement thickness, as shown in Fig.16.

Some uncertainty exists in the curves shown in Figs.15-16 due mainly to the lack of data close to the blade surface where some rough assumptions have been made regarding the velocity profiles. Boundary layer Pitot probe traverse data, shown in Fig.17, improved the situation in the region of high turbulence close to the blade surface. But, since the flow in the interaction region can no longer be regarded as steady, and since reversed flow does occur, the readings are erroneous, particularly close to the wall, so that the integral parameters are approximations showing only a trend rather than absolute values.

When the integral parameters obtained from the L2F surveys (Figs.15-16) and those of the boundary layer probe traverse, shown in Fig.18, are compared, the observed trend of the AVDR influence on the boundary layer loading, and consequently on the viscous losses, is confirmed. At the low AVDR value (1.01) for the probe traverse data (compared to AVDR values of 1.07 and 1.13 for the L2F data), the suction surface boundary layer showed a further increase in the displacement thickness ($\Delta \delta / \delta = 10$), as well as a further increase in the maximum value of shape factor $H_{12}$ to about 5.3.
CONCLUSIONS

The interaction of a strong shock-wave with the turbulent, blade suction surface boundary layer of a supersonic compressor cascade has been investigated experimentally. At an inlet Mach number of 1.5 the cascade flow was throttled to achieve a static pressure ratio of 2.5. The first blade-passage shock-wave with pre-shock Mach numbers near the blade surface of 1.44 to 1.52, induced a strong boundary layer separation at about 64 to 65 percent of chord.

The results showed that behind the passage shock-waves a relatively strong adverse pressure gradient is imposed on the separated boundary layer. Oil streak lines and ink injections indicated that for the investigated flow conditions, there is no boundary layer reattachment on the rear part of the blade suction surface.

Furthermore, it was found that the axial blade passage stream sheet contraction - expressed by the AVDR - has a significant influence on the strong shock/boundary layer interaction and the resulting total pressure losses. High AVDR values seem to reduce the pressure gradient across the interaction region, and thereby reduce shock losses and viscous losses from boundary layer separation.

In contrast to earlier shock/boundary layer interaction experiments on flat plates, the local flow field behind the lambda and quasi-normal shock system did not show the so-called "supersonic tongue". In the present experiments a small local subsonic flow region was observed immediately behind the quasi-normal shocks of the bifurcated shock system, so only the flow near the outer edge of the boundary layer remained supersonic throughout the whole interaction region.

REFERENCE


APPENDIX

Cascade Geometry

<table>
<thead>
<tr>
<th>blade chord</th>
<th>t/c</th>
<th>0.65</th>
</tr>
</thead>
<tbody>
<tr>
<td>gap/chord ratio</td>
<td>c</td>
<td>170 mm</td>
</tr>
<tr>
<td>stagger angle</td>
<td>βs</td>
<td>148.1°</td>
</tr>
</tbody>
</table>

Coordinates of the manufactured and tested blade:

<table>
<thead>
<tr>
<th>suction surface</th>
<th>pressure surface</th>
</tr>
</thead>
<tbody>
<tr>
<td>XS/C</td>
<td>YS/C</td>
</tr>
<tr>
<td>0.005340</td>
<td>0.000015</td>
</tr>
<tr>
<td>0.006305</td>
<td>0.006805</td>
</tr>
<tr>
<td>0.012863</td>
<td>0.006423</td>
</tr>
<tr>
<td>0.037199</td>
<td>0.005176</td>
</tr>
<tr>
<td>0.072572</td>
<td>0.003724</td>
</tr>
<tr>
<td>0.107956</td>
<td>0.002772</td>
</tr>
<tr>
<td>0.143372</td>
<td>0.002416</td>
</tr>
<tr>
<td>0.178825</td>
<td>0.003246</td>
</tr>
<tr>
<td>0.214218</td>
<td>0.005099</td>
</tr>
<tr>
<td>0.249591</td>
<td>0.007495</td>
</tr>
<tr>
<td>0.284948</td>
<td>0.010173</td>
</tr>
<tr>
<td>0.320298</td>
<td>0.012964</td>
</tr>
<tr>
<td>0.355636</td>
<td>0.015781</td>
</tr>
<tr>
<td>0.390966</td>
<td>0.018516</td>
</tr>
<tr>
<td>0.426311</td>
<td>0.021374</td>
</tr>
<tr>
<td>0.461655</td>
<td>0.024133</td>
</tr>
<tr>
<td>0.496983</td>
<td>0.026848</td>
</tr>
<tr>
<td>0.532274</td>
<td>0.029115</td>
</tr>
<tr>
<td>0.567538</td>
<td>0.030781</td>
</tr>
<tr>
<td>0.602771</td>
<td>0.031658</td>
</tr>
<tr>
<td>0.637364</td>
<td>0.031709</td>
</tr>
<tr>
<td>0.666843</td>
<td>0.030636</td>
</tr>
<tr>
<td>0.704111</td>
<td>0.028336</td>
</tr>
<tr>
<td>0.739401</td>
<td>0.025599</td>
</tr>
<tr>
<td>0.774711</td>
<td>0.022500</td>
</tr>
<tr>
<td>0.810045</td>
<td>0.019415</td>
</tr>
<tr>
<td>0.845370</td>
<td>0.016439</td>
</tr>
<tr>
<td>0.880703</td>
<td>0.013429</td>
</tr>
<tr>
<td>0.916019</td>
<td>0.010341</td>
</tr>
<tr>
<td>0.951345</td>
<td>0.007626</td>
</tr>
<tr>
<td>0.973404</td>
<td>0.005762</td>
</tr>
<tr>
<td>0.996600</td>
<td>0.003798</td>
</tr>
</tbody>
</table>