Use of Supersonic Cascades made of Blades of Simple Geometric Shapes for Cascade Wind Tunnel Performance Evaluation

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Use of blades of simple geometric shape for supersonic compressor cascade tests gives easy means to check the validity of wind tunnel tests, since comparison of experimental and theoretical shock and flow patterns as well as pressure distributions is simplified due to the fact that there exists only a limited number of discrete shock waves or expansion fans. A supersonic compressor blade cascade wind tunnel performances are evaluated according to this technique.

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INTRODUCTION

Performances of compressor blade cascades obtained in supersonic wind tunnels are strongly influenced by the axial velocity ratio due namely to wall boundary layer growth. In order to control the validity of experimental results, series of tests were performed on blade cascades for which pressure distribution and shock wave pattern were well known. Blades of simple geometric shape were chosen for this experimentation, such as triangular or slightly more complicated polygonal blades. In such cascades, the shock waves and expansion fans induced by the change in direction of the blade surface form discrete series of waves. Comparison between theory and experiment is easy. Each shock wave or expansion fan shown by the Schlieren picture of the flow field can be compared to the corresponding theoretical one.

Another check of test results validity is obtained by comparing theoretical and experimental pressure distributions along the blade mid-span. Experiment confirms the theoretical predictions as long as not too severe boundary layer separations appear.

NOMENCLATURE

\( i \) = incidence angle, angle of undisturbed flow and cascade inlet plane
\( M \) = actual Mach number of undisturbed flow
\( M_c \) = Mach number at cascade inlet
\( M_d \) = wind tunnel design Mach number
\( p_r \) = upstream total pressure
\( p_a \) = upstream static pressure
\( p \) = local static pressure
\( q = \frac{1}{2} \gamma M^2 p_a \) = upstream dynamic head
\( \beta \) = cascade setting angle, angle of undisturbed flow and cascade inlet plane
\( \gamma \) = isentropic coefficient
\( \sigma \) = solidity
\( \Delta p_i \) = total pressure loss.

The experimental investigation performed in order to give a guidance for experiments to be made with blade cascades, the theoretical flow field of which is not known, is described in this paper.

TEST INSTALLATION

A schematic representation of the test facility and details of the test chamber are given in Fig.1. The air supply is furnished partly by a continuous 14 atm, 10 kg/sec primary flow that drives an additional 3 kg/sec atmospheric pressure secondary flow. A mixing chamber with screens uniformizes the air supply and delivers a 2.5 atm, 13 kg/sec flow in the test chamber. Due to heating of the primary air during compression, the tests were performed at stagnation temperature be-
between 350 and 370 K, thus limiting the difficulties due to condensation shocks.

The wind tunnel is made with two parallel side walls and shaped top and bottom walls. A uniform supersonic flow is delivered at cascade inlet. Wind tunnel geometric and aerodynamic data are given in Table 1.

Table 1 Supersonic Compressor
Blade Cascade Wind Tunnel

<p>| | |</p>
<table>
<thead>
<tr>
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</tr>
</thead>
<tbody>
<tr>
<td><strong>Air supply</strong></td>
<td></td>
</tr>
<tr>
<td>Mass flow rate</td>
<td>13 kg/sec</td>
</tr>
<tr>
<td>Stagnation pressure</td>
<td>2.3 atm</td>
</tr>
<tr>
<td>Stagnation temperature</td>
<td>350 - 370 K</td>
</tr>
<tr>
<td><strong>Test chamber geometry</strong></td>
<td></td>
</tr>
<tr>
<td>Height</td>
<td>247 mm</td>
</tr>
<tr>
<td>Width</td>
<td>100 mm</td>
</tr>
<tr>
<td>Window diameter</td>
<td>336 mm</td>
</tr>
<tr>
<td><strong>Test Mach number</strong></td>
<td></td>
</tr>
<tr>
<td>Design</td>
<td>1.5, 1.75, 2</td>
</tr>
<tr>
<td>Actual</td>
<td>1.38, 1.56, 1.81</td>
</tr>
</tbody>
</table>

*Corresponds to undisturbed upstream flow. At cascade inlet may be greater or smaller according to blade stagger angle and the corresponding expansion or shock wave from the most upstream blade in the cascade.*

Test chamber side walls are made of circular plates on which the blades are fixed. Setting angle of cascade, \( \beta \), defined as the angle of the normal to the cascade inlet plane with the undisturbed upstream velocity can be modified by means of rotating the wall plates. Blade stagger can be changed by means of changing wall plates only.

Two types of wall plates are used:

1. Large, circular plexiglass plates for Schlieren picture visualization, Fig.2(a). The optical quality of these plates is good enough to give the details of shock waves and boundary layer separation on blade surfaces. Due to the large size of these windows, the shock pattern well upstream and downstream of the cascade can be visualized.

2. Metal plates for flow parameter measurements (total and static pressure, flow angle). A plexiglass insert allows shock wave pattern visualization in and immediately upstream and downstream of the cascade. The pressure taps and traverses, represented schematically in Fig.1(b), are actually not bored on the same side of the wind tunnel. Pressure traverses for upstream flow calibration (that is made before each test) and movable downstream probe traverses for flow angle and pressure loss measurements are on one side of the wind tunnel. Wall pressure taps are on the other side, Fig.2(b).

Total and static pressure measurements made upstream of the cascade show that:

1. Boundary layers on wind tunnel side walls have no negligible thickness.
2. Humidity of the air cannot be neglected. Experiments are made only when wind tunnel air has attained sufficiently high temperature.
3. Actual Mach number of the undisturbed upstream flow has to be determined before each run.

Starting or choking of blade passages is obtained by means of a downstream throttle valve and also with the help of two flaps situated at cascade top and bottom downstream ends. These flaps also define the cascade outlet flow direction.

Downstream flow exploration is made by means of various conventional double probes (total head, static pressures, and flow angle) introduced through a slot parallel to the cascade outlet plane. Measurements can be made at various distances from the sidewalls. Systematic deflection and loss coefficient measurements are made at blade mid-span. Usually three blade channels are explored, in order to check flow periodicity.

**TRIANGULAR AND TRAPEZOIDAL BLADES**

Some preliminary investigation was made with thick (9 percent) triangular blades [Table 2, column (a)] at design Mach number 1.5. A supersonic configuration was obtained only with low solidity \((C = 0.75)\) and at \(5^\circ\) deg incidence. No deflection of the flow was observed.

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Fig. 3 Shock and flow pattern:
Triangular blades ($M = 1.5$)

Fig. 4 Shock pattern:
Trapezoidal blades ($M = 1.5$)

Fig. 5 Shock pattern and pressure distributions:
double triangular blades DT 60 - 6 - 18 ($M_d = 1.75$)
Setting angle 57.5°

Fig. 3 compares the theoretical flow and shock pattern, obtained by means of the graphical method of characteristics, with the experimental results. The similarity between theoretical and experimental shock waves is satisfactory. The curvature of the leading edge shock wave, due to the interaction with the apex expansion fan, is clearly visible.

A unique, unexpected water injection into the wind tunnel, due to some defect in the air supply pipes, allowed wall streamline visualization. The visualized wall streamlines are much similar to the theoretical ones, but they undergo a change in direction at some distance ahead of the shock waves. This phenomenon is explained by shock wave/wall boundary layer interaction. It emphasizes the necessity of pressure measurements at mid-span of the blades and not along the wind tunnel walls.

When the blade apexes were cut off to form a thinner (5 percent) trapezoidal blade [Table 2, column (b)], the cascade could be started with a higher solidity ($S = 2$). Fig. 4 indicates a satisfactory comparison between theory and experiment over most of the blade length, but boundary layer
Fig. 6 Shock pattern and pressure distributions: double triangular blades DT 60 - 6 - 18 ($M_d = 1.75$) Setting angle 60°

Fig. 7 Shock pattern and pressure distributions: double triangular blades DT 60 - 6 - 18 ($M_d = 1.75$) Setting angle 62.5°

Table 2 Geometric Definition of the Blades

<table>
<thead>
<tr>
<th>Blade type</th>
<th>(a) Triangular</th>
<th>(b) Trapezoidal</th>
<th>(c) Double Triangular</th>
<th>(d) Flat Perforation</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chord (mm)</td>
<td>60</td>
<td>60</td>
<td>60</td>
<td>88.6</td>
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<tr>
<td>Maximum thickness (mm)</td>
<td>5.2</td>
<td>3</td>
<td>3</td>
<td>11</td>
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<tr>
<td>Leading edge angle (°)</td>
<td>10</td>
<td>10</td>
<td>6</td>
<td>10</td>
</tr>
<tr>
<td>Apex angle (°)</td>
<td>20</td>
<td>-</td>
<td>18</td>
<td>-</td>
</tr>
<tr>
<td>Number of blades</td>
<td>4</td>
<td>5</td>
<td>6</td>
<td>9</td>
</tr>
<tr>
<td>Solidity</td>
<td>0.75</td>
<td>2</td>
<td>2</td>
<td>2.90</td>
</tr>
<tr>
<td>Cascade setting angle (°)</td>
<td>65</td>
<td>65</td>
<td>57.5 60 62.5</td>
<td>60</td>
</tr>
<tr>
<td>Incidence (*) angle (°)</td>
<td>5</td>
<td>5</td>
<td>2.5 0 2.5</td>
<td>0</td>
</tr>
<tr>
<td>Test design Mach number</td>
<td>1.5</td>
<td>1.5</td>
<td>1.5 1.75</td>
<td>2</td>
</tr>
</tbody>
</table>

* Incidence angle is defined as the angle between undisturbed upstream velocity and flat upper surface of the blades.
separation appears at the junction of the trailing edge shock wave and the adjacent blade inner surface.

DOUBLE-TRIANGULAR BLADES

In order to induce a more noticeable deflection of the flow at cascade outlet, double-triangular blades [Table 2, column (c)] were tested at high solidity (\(\sigma = 2\)). Figs. 5 to 7 show the role of cascade setting angle that varies from 57.5 to 62.5 deg. As cascade setting angle increases, blade upper surface pressure decreases; i.e., inlet Mach number increases. This is due to the "unique incidence" imposed by the most upstream blade of the cascade. Table 3 gives the correspondence between wind tunnel design Mach number, \(M_d\), actual Mach number of the undisturbed upstream flow, \(M_a\), and cascade inlet Mach number, \(M_c\).

Table 3 Influence of Cascade Setting Angle on Upstream Flow Characteristics

<table>
<thead>
<tr>
<th>Figure</th>
<th>5</th>
<th>6</th>
<th>7</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cascade setting angle (deg)</td>
<td>57.5</td>
<td>60</td>
<td>62.5</td>
</tr>
<tr>
<td>Design Mach number, (M_d)</td>
<td>1.75</td>
<td>1.75</td>
<td>1.75</td>
</tr>
<tr>
<td>Undisturbed actual Mach number, (M_a)</td>
<td>1.56</td>
<td>1.56</td>
<td>1.56</td>
</tr>
<tr>
<td>Cascade inlet Mach number, (M_c)</td>
<td>1.495</td>
<td>1.56</td>
<td>1.735</td>
</tr>
</tbody>
</table>

Existence of a shock wave on blade leading edge, while theory suggests a complete absence of perturbation, is a very usual phenomenon due to the finite thickness of blade leading edge. But one notices that the shock wave induced by the most upstream blade of the cascade in Fig.5, becomes weak in Fig.6, where it should theoretically vanish, and is still weaker in Fig.7 where expansion waves, issued from the front leading edge, decrease the strength of the shock wave due to leading edge thickness.

Pressure distributions measured at mid-span of the blades correspond reasonably well to the expected theoretical pressure distributions. On the upper surface, the pressure curve is interpreted as made up by three constant pressure lines: the first one corresponds to the flat entrance of the upper surface, the second to the pressure behind the apex expansion wave, and the third to the pressure behind the point of junction with the shock wave issuing from the leading edge of the adjacent blade.

A more complicated pressure distribution is induced on the blade inner face. It is due to interaction of the apex expansion wave with the wall, to the shock wave induced at the point of change in direction of the blade surface, and to the reflection of the leading edge shock wave that already hit the adjacent blade.

The complicated pressure distributions indicated by the calculated values could not have been guessed from the insufficient number of pressure recordings.

Similar results are obtained with the other wind tunnel nozzles. An example is given for wind tunnel design Mach number 1.5 in Fig.8. In all these cases, the trailing edge shock wave gives an inverse flow deflection that seriously diminishes the deflection performances of the cascade. This result is verified by flow angle measurements by means of probes and Schlieren picture wake visualization.


HIGH DEFLECTION BLADES

Series of polygonal blades designed for high...
deflection \([\text{Table 2, column (d)}]\) were tested at design Mach number 2.

The cascade solidity was high in order to obtain the design flow deflection (solidity \(S = 2.9\); deflection 50 deg). Due to the high value of the actual Mach number at cascade inlet \((M_a = M_c = 1.81)\), starting of the blade passages was possible.

Comparison of theoretical and experimental shock patterns is satisfactory (Fig.9). The various shock and expansion fan reflections predicted by theory can be found on the actual Schlieren picture.

\[ M = 2 \quad \beta = 60^\circ \]
OFF-DESIGN OPERATION OF POLYGONAL BLADES

It can be noticed on the Schlieren pictures, presented in this paper, that there is no important flow separation on the blades as long as the trailing edge shock wave does not hit the adjacent blade inner surface. Though there is no boundary layer suction, the shocks issued from side walls and due to boundary layer growth do not seem to affect the test section at mid-span. In this two-dimensional case, a deflection of the flow toward a direction normal to the cascade inlet plane results in an expansion of the flow (increase of Mach number and decrease of pressure) that may explain the non-separation of the flow.

When the back pressure is increased, flow configurations are obtained that cannot be compared to simple theoretical predictions. Fig.10 shows the choking of the trapezoidal blade cascade of Fig.4. At low back pressure, the blade channels are started, Fig.10(a). When downstream pressure is increased, choking is obtained with a set of reflected shock waves in periodic arrangement, Fig.10(b).

PRESSURE LOSSES

The main difference between wind tunnel tests and theoretical calculations consists in pressure losses. Results of probe traverses downstream of the double triangular blade cascade of Fig.6 are shown in Fig.11 as examples of pressure losses and actual deflection angles. In the case of started blade passages, Fig.11(a), the pressure losses are mainly limited to the wakes of the blades. The higher losses on the left part of Fig.11(a) are due to probe interactions. Flow deflection is nearly constant.

When the blade channels are choked, the mean level of pressure losses is higher, Fig.11(b). The peak pressure loss, measured in the wakes, is almost the same in the case of started and unstarted channels. Deflection of the flow is also the same in both cases.

CONCLUSION

Use of supersonic compressor blade cascades made of polygonal blades of simple shape provides a satisfactory method for checking the validity of wind tunnel measurements.

In the wind tunnel used for these tests, a high span-to-chord ratio of 1.66 gives an essentially undisturbed operation at the mid-span test section. Experimental pressure distributions on blade upper and inner surface checked favorably with predictions obtained by means of a graphical method of characteristics.