Pressure distribution on a biconvex airfoil in supersonic flow

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1 Summary

The aim of the laboratory experiment is to determine the pressure distribution on a symmetrical biconvex 2-D airfoil made of two circular arcs. Each lab group runs the experiment at a unique angle of attack and free stream Mach number. Theoretical predictions obtained by shock-expansion and linear approximation are compared with the experimentally determined pressure coefficient.

2 Experimental rig and test equipment

The experiment is run in the supersonic wind tunnel, VM100, at Heat and Power Technology, for the location see the map on the last page. The tunnel is supplied with air from a two stage screw compressor (ATLAS COPCO) \( P_{max} = 1.3\text{MW} \) at \( 4.7\text{kg/s}, 400\text{kPa} \). The hot compressed air is cooled from \( 180^\circ\text{C} \) to room temperature in a heat exchanger before entering the wind tunnel.

![Wind tunnel layout](image)

Figure 1: Wind tunnel layout. a) Stagnation chamber; b) Nozzle; c) Test section; d) Diffuser
1) Flow rectifier; 2) Screen; 3) Jack (mechanism to adjust the critical section); 4) Plate support; 5) Flexible plate (spring blade)

The wind tunnel, see figure 1, consists of the following main parts, starting at the inlet: A stagnation-chamber with rectifier and screens, a convergent divergent 2-D nozzle, a test-section
and finally a diffuser connected to an outlet channel with sound absorbing walls. The flow in the nozzle, outside the boundary layers, is isentropic all the way down to the test section (cross section: $10 \times 10 \text{ cm}^2$) where the Mach number is constant, $M_\infty$, and the flow is parallel. $M_\infty$ is determined solely by the area ratio of the test section to the critical section. This simple relation holds for supersonic flow of an ideal gas with constant $\gamma$. Supersonic flow in the channel can only be maintained as long as the pressure in the stagnation chamber is high enough to compensate for downstream pressure losses. When the Mach number is increased the stagnation chamber pressure has to be increased since the downstream losses increase sharply with increasing Mach number.

The tunnel is designed in such a way that $M_\infty$ can be changed while the tunnel is running. The horizontal walls in the 2D nozzle between the critical section and the test section are made of two ingeniously designed thin steel plates or spring blades. When the critical section is changed the flexible walls change their curvature such that the nozzle is maintained shock-free. To simplify the construction the nozzle has to be relatively long which however also causes a thick (undesirable) boundary layer in the test section.

![Diagram](image)

Figure 2: Details of the airfoil, location of pressure taps

The model in the test-section is a symmetric biconvex 2D airfoil with a span equal to the wall distance, see figure 2. The airfoil has seven pressure taps in the lower surface and one in the upper surface. The pressure tap in the upper surface is at the same position relative to the leading edge as the first pressure tap in the lower surface.

The model is mounted on a strut-sting combination, which is fixed to a sword. The sword passes through the wind tunnel floor and ends in the model-positioning device. The sword is shaped as a circular arc and its center coincide with the symmetry point of the airfoil, see figure 3.

The angle of attack ($\alpha$) is changed by turning the sword around its axis and the change is readily read out on a scale on the positioning device. However, $\alpha$ has to be corrected for a zero point error caused by the deflection of the model due to unsymmetrical aerodynamic loads during the run. The true zero of attack is found by adjusting the sword until the two symmetrically positioned pressure taps give the same value.

The pressure taps on the model are connected to electric pressure sensors via tubes inside the sword. A data acquisition system (computer with AD converter) connected to the sensors shows
Figure 3: Close up view of test section and positioning device. 1) Biconvex airfoil; 2) Strut; 3) Sting; 4) Sword; 5) Read out for the angle of attack; 6) Pressure tubes connected to the model

the pressure both in an analogue and digital fashion on the monitor screen. The pressure readings are also saved as a file in the computer for further evaluation. In addition to the pressures on the model also the pressure in the stagnation chamber is registered.

3 The experiment

The result from the pressure measurements is presented in a set of two curves for each value of $M_\infty$ and $\alpha$, one for the upper side and one for the lower side of the model. The curves show the pressure coefficient $c_p$ as a function of the normalized position $(x/c)$ (where $c$ is the chord) on the model surface. Since the profile is symmetric, pressure holes are only needed on one side, in our case the lower side. To measure on the other side we make a second run with the opposite sign of the angle of attack. The definition of $c_p$ is:

$$c_p = \frac{p - p_\infty}{q_\infty}.$$

It can be rewritten in the following way (show this):

$$c_p = \frac{2}{\gamma M_\infty^2} \left( \frac{p}{p_\infty} - 1 \right).$$
\( p_\infty \) is not measured directly but is calculated from the stagnation pressure and \( M_\infty \), by assuming isentropic flow between the stagnation chamber and the test section. The single pressure tap on the upper side is not used in the calculations but is registered and used as a check on the angle of attack when changing from positive to negative \( \alpha \).

4 Determination of the theoretical pressure distribution

The pressure coefficient curves from the experiment will be compared with the theoretically determined distributions. Two theoretical methods should be used: The linear approximation and the shock-expansion approximation. In both cases the calculations should be carried out with 4 significant digits. In the linear approximation, the pressure coefficient in a point on the surface is simply a function of the slope of the surface relative the free stream (\( \theta \)) and \( M_\infty \):

\[
c_p = \frac{2\theta}{\sqrt{M_\infty^2 - 1}}
\]

If the angle between the surface and cord is denoted \( \varphi \) (\( \varphi > 0 \) for \( x < c/2 \) and \( \varphi < 0 \) for \( x > c/2 \)) we have:
- upper surface: \( \theta(x) = \varphi(x) - \alpha \)
- lower surface: \( \theta(x) = \varphi(x) + \alpha \)

\( \varphi(x) \) is calculated from the geometry in figure 4.

In the shock-expansion approximation, the condition downstream of the attached oblique shock is first calculated and then the surface Mach number is determined using the Prandtl-Mayer function. Shock and isentropic relations give the pressure relation \( p/p_\infty \) expressed as a pressure coefficient. For the calculations it is convenient to use the Compressible Aerodynamics Calculator: http://www.dept.aoe.vt.edu/devenpor/aoe3114/calc.html
**Homework done before the lab:** Determine the pressure coefficient along the profile on both the upper and lower sides, using both the linear method and the shock-expansion method. Plot your determined distributions in the diagram found on the last page. **Do not use your own diagram.**

## 5 Errors

A few reasons for the discrepancy between experiment and theory are stated below:

### 5.1

Both methods are approximations. In the shock-expansion approximation the disturbance in surface pressure from outgoing Mach-waves reflected back by the shock is omitted and the flow is regarded as a simple wave flow. In the linear theory shocks are neglected and the flow is considered isentropic.

### 5.2

In the calculations the boundary layer on the surface is not taken into consideration. This approximation is justified in the leading part of the boundary layer but not in the rear part where shock induced separation will cause large deviations between calculated and experimentally obtained pressure.

### 5.3

Ideally the strut generates disturbances only propagating down-stream in a supersonic flow. However, pressure fluctuations from the strut also propagate upstream in the subsonic boundary layer. The strut may even affect the pressure in the rear holes on the opposite side as the strut influences trailing edge boundary layer separation.

### 5.4

The leading edge is slightly rounded. The ideally straight attached shock is locally a curved detached shock with local subsonic flow. This influence is minor, and the deviation in shock angle from the calculated one is small at the position of the first hole.
Figure 5: Map showing where to find the VM 100 wind tunnel on Campus
Figure 6: Plot your data here