# 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 AOA and freestream 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 *VM100 Transonic Wind Tunnel*, at the division of Heat and Power Technology, Department of Energy Technology (for the location, see the map on page 7). The tunnel is supplied with air from a two stage screw compressor (ATLAS COPCO, $P_{max} = 1.3MW$ at $4.7kg/s, 400kPa$). The hot compressed air is cooled from $180°C$ to room temperature in a heat exchanger before entering the wind tunnel.

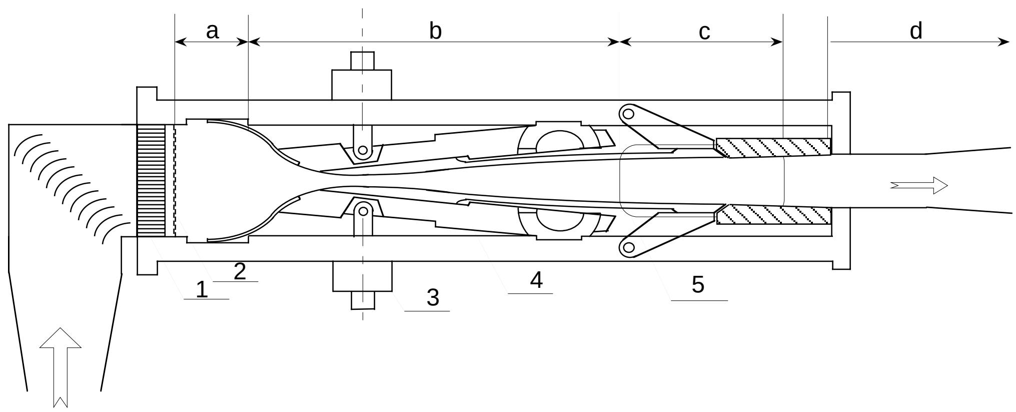


Figure 1: Wind tunnel layout. a) Stagnation chamber; b) Nozzle; c) Test section; d) Diffuser 1) Flow rectifier; 2) Screen; 3) Jack (mechanism to adjust the critical section); 4) Plate support; 5) Flexible plate (spring blade)

The wind tunnel, described in 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: $10 \times 10$ 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.

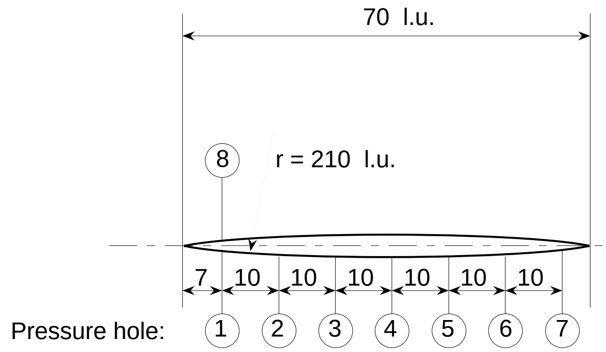


Figure 2: Details of the airfoil, location of pressure taps

The model in the test-section is a symmetric biconvex 2D airfoil with a span equal to the wall distance as shown in 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 in the shape of a circular arc and its center coincide with the symmetry point of the airfoil, as seen in Figure 3.

The angle of attack (AOA), $\alpha$, is changed by turning the sword around its axis and the change is readily read out on a scale on the positioning device. However, the AOA 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 AOA is found by adjusting the sword until the two symmetrically positioned pressure taps give the same value.

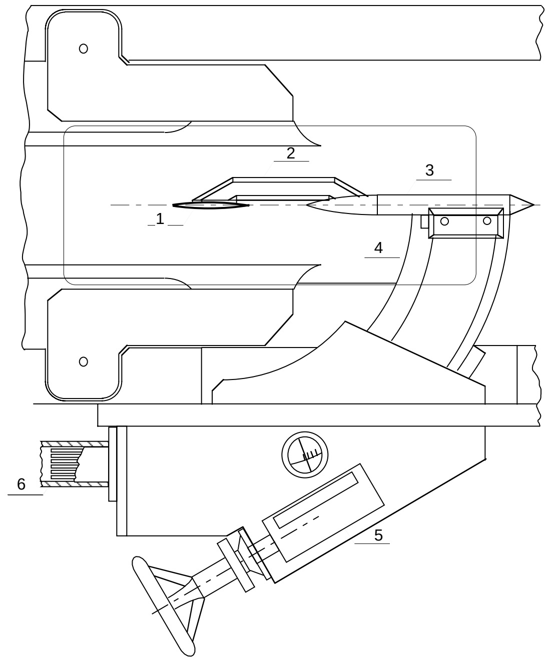


Figure 3: Close up view of test section and positioning device. 1) Biconvex airfoil; 2) Strut; 3) Sting; 4) Sword; 5) Read out for the angle of attack; 6) Pressure tubes connected to the model

The pressure 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 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.

For demonstration purpose, the Schlieren technique is used to visualise the flow around the airfoil throughout the experiment. Its schematic is shown in Figure 4. The only difference in setup compared to Shadowgraphy (used in the laboratory exercise: Flow in a shock tube) is the addition of the knife edge. This makes it possible to have a focused image and quantitatively analyse the acquired image, unlike in the case of Shadowgraphy. More details are provided in Refs. [1, 2] for interested students: uploaded on Canvas as "Merzkirch-Egami_2007.pdf" and "Settles_2001.pdf".

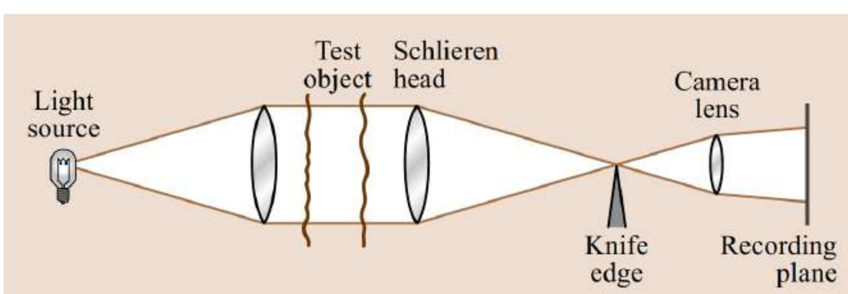


Figure 4: Schematic of Schlieren setup [1]

## 3 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 surface and one for the lower surface of the model. The curves show the pressure coefficient, $c_p$, as a function of the normalised position, $x/c$ (where $c$ is the chord), on the model surface. Since the airfoil profile is symmetric, pressure holes are only needed on one side, in our case the lower side. To measure on the other side, which is upper side in our case, we make a second run with the opposite sign of the AOA. Combining the two, we obtain a full $c_p$ profile for a positive AOA.
The definition of $c_p$ is:

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

It can be rewritten in the following way:

$$c_p = \frac{2}{\gamma M_\infty^2}\left(\frac{p}{p_\infty} - 1\right).$$

The static freestream pressure, $p_\infty$, 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 AOA when changing from a positive to a 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 to the freestream, $\theta$, and $M_\infty$:

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

If the angle between the surface and chord is denoted $\varphi$ ($\varphi > 0$ for $x < c/2$ and $\varphi < 0$ for $x > c/2$), we have:

$$\text{upper surface: } \theta(x) = \varphi(x) - \alpha,$$
$$\text{lower surface: } \theta(x) = \varphi(x) + \alpha,$$

where, $\varphi(x)$ is calculated from the geometry in Figure 5.

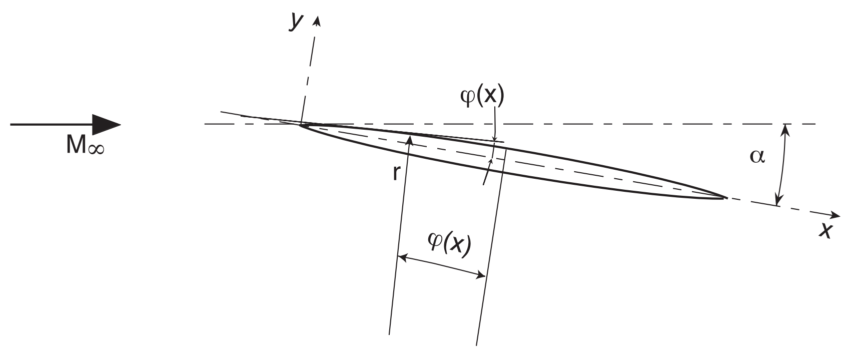


Figure 5: Definition of angles

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, if you are not using a programming code such as Matlab, it is convenient to use the *Compressible Aerodynamics Calculator* available at:
`http://www.dept.aoe.vt.edu/~devenpor/aoe3114/calc.html`

# 5 Evaluation of data

**Homework done before the laboratory session**:

Determine the pressure coefficient along the airfoil on both the upper and lower surfaces, using both the linear method and the shock-expansion method. Plot your determined distributions in a diagram, ready to be compared against the experimental data. Using a programming code such as Matlab is recommended, but if you choose not to, then the diagram attached on the last page could be used. Bring your laptop/calculator to the laboratory session.

**Analysis session during the laboratory session**:

Plot the pressure coefficient profile from the obtained experimental data and compare it against the theoretical curves. How well do they match?

# 6 Errors

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

## 6.1 Approximations

Both methods are approximations. In the shock-expansion approximation the disturbance in the surface pressure from the 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.

## 6.2 Shock boundary layer interaction at the trailing edge

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 values.

## 6.3 Disturbances generated by the strut-sting structure

Ideally, the strut generates disturbances only propagating downstream 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.

## 6.4 Effects of a rounded leading edge

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.

**REFERENCES**

[1] Tropea C, Yarin AL, Foss JF. Springer handbook of experimental fluid mechanics. Springer; 2007.

[2] Settles GS. Schlieren and shadowgraph techniques. Springer; 2001.


Figure 6: Map showing where to find the VM 100 wind tunnel on Campus

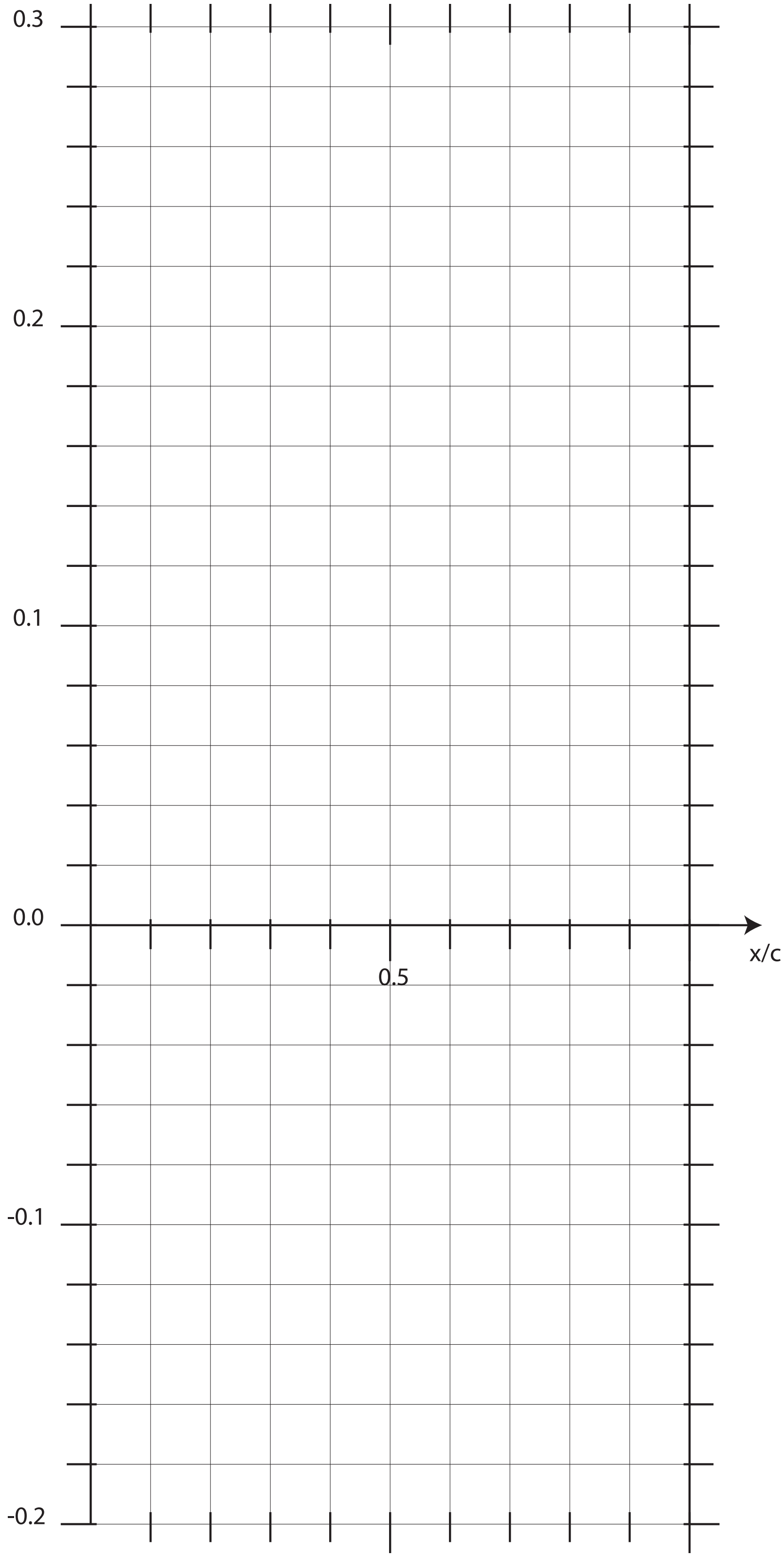


Figure 7: Plot your data here