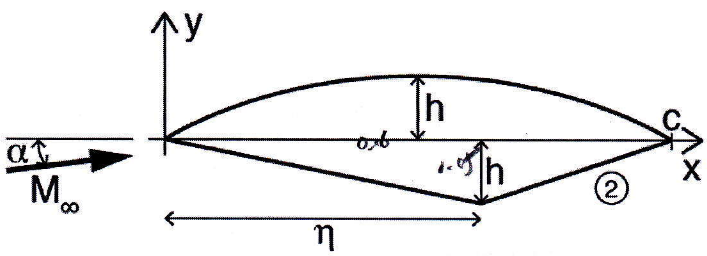
Delft University of Technology **DEPARTMENT OF AEROSPACE ENGINEERING**		
Course: Aerodynamics 2;	Code: AE2130-III	Course year: 2
Date: Tuesday 28 January 2014		Time: 14:00 – 17:00

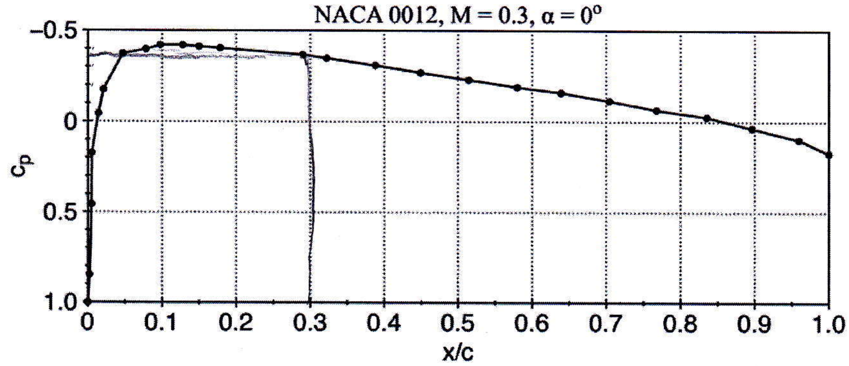
On the top of <u>each</u> answer sheet write: initials, name, student number, sheet number/total number of sheets
This exam consists of 5 questions.

Problem 1 [25 points]
An airfoil is immersed in a supersonic flow at a free stream Mach number $M_\infty = 3$. The geometry of the upper side is given by the equation $\frac{y}{c} = \frac{h}{c}\sin\left(\frac{\pi x}{c}\right)$. The lower side has two straight sections. The angle of attack is $\alpha = 3^o$, $\frac{h}{c} = 0.05$ and $\frac{\eta}{c} = 0.6$. You may use linear theory to answer the following questions.



i. Under what assumptions can linear theory be applied?
ii. Compute the pressure coefficient for the lower and upper side as function of x/c. Plot the result in a diagram. For which values of x/c (if any) is the pressure coefficient equal to 0?
iii. Use shock-expansion theory to compute the pressure coefficient in region 2 and compare its value with that resulting from linear theory.

Problem 2 [20 points]
Consider a NACA 0012 airfoil, the distribution of the pressure coefficient on the upper side for $M = 0.3$ at $\alpha = 0^o$ is shown below:



i. Determine the pressure coefficient at $x/c = 0.3$ for $M = 0.75$.
ii. What is the value of the critical Mach number for this airfoil at $\alpha = 0^o$?
iii. Explain what is the effect of making the airfoil thicker in terms of critical Mach number.

Problem 3 [25 points]

A rocket engine is installed in a test stand at sea level ($p_{amb} = 1 \times 10^5\ Pa$). In the combustion chamber of the engine the total temperature is $T_o = 3000\ ^oC$. The throat of the engine has a diameter of $0.1\ m$ and the mass-flow through the engine is $\dot{m} = 20\ kg/s$. The Mach number at the exit is $M = 4$. Since the engine burns LOX with LH2, the gas constant of the combustion gasses is $R = 594\ J/(kg\ K)$ and $\gamma = 1.2$.

i. Start from the continuity equation and show that for choked flow the mass flow through the engine can be computed as:

$$\dot{m} = \frac{p_0 A^*}{\sqrt{T_0}} \sqrt{\frac{\gamma}{R}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}$$

ii. Determine the total pressure in the combustion chamber, the nozzle exit cross section and the velocity at the nozzle exit.

iii. Indicate for which range of combustion chamber pressures (give the values) the flow through the nozzle is (a) underexpanded and (b) overexpanded

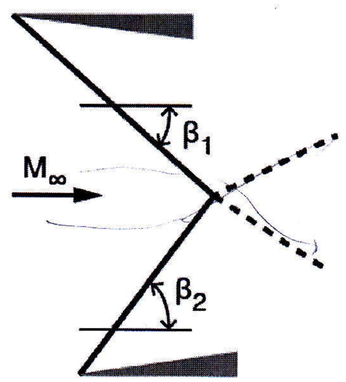
Problem 4 [20 points]

i. Consider a stationary oblique shock wave in a supersonic flow. Indicate what happens to the following 4 variables when crossing the shock (stay constant, increase or decrease), justify you answer: (a) tangential Mach number, (b) kinetic energy, (c) Mach angle, (d) total density.

ii. Sketch the graph expressing the M-β-θ relationship for an oblique shockwave and discuss it. In your discussion include the following concepts: maximum deflection angle, normal shock wave solution, strong solution, weak solution and Mach angle.

Problem 5 [10 points]

Consider two shocks having arbitrary shock angles β_1 and β_2. The shocks intersect in a point (see figure on the right). As a result two new waves are formed. Explain wether these are expansion or compressions waves.



Values of gas properties

Universal gas constant: $R_0 = 8314$ J/Kmol K; Air gas constant: $R_{air} = 287$ J/Kg K;
Specific heat of air: $C_p = 1004$ J/Kg K