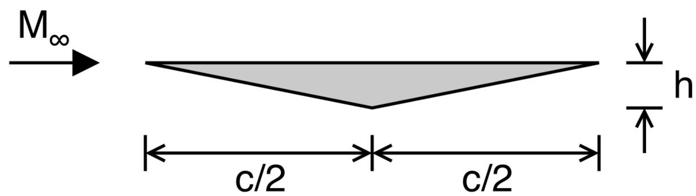
Delft University of Technology **DEPARTMENT OF AEROSPACE ENGINEERING**	
Course: Thermodynamics and compressible aerodynamics; Code AE2-125 **Aerodynamics only; exam for students that already passed the thermodynamics test held in April 2010**	Course year: 2
Date: Friday 25th June 2010	Time: 14 – 16

Problem 1-a

An airfoil is schematically represented in the figure below, it is immersed in a supersonic flow at $M_\infty = 3$. The airfoil has a thickness of $h/c = 0.12$, where c is the airfoil chord.



(i) Sketch the following flow field features: shock waves, expansion waves, streamlines and slip-lines.
(ii) Determine the pressure distribution and lift coefficient according to linearized theory.
(iii) Determine the pressure distribution and lift coefficient according to shock-expansion theory.
(iv) Compare the results computed in (ii) and (iii) using in the discussion the hypotheses that are made in applying linearized theory

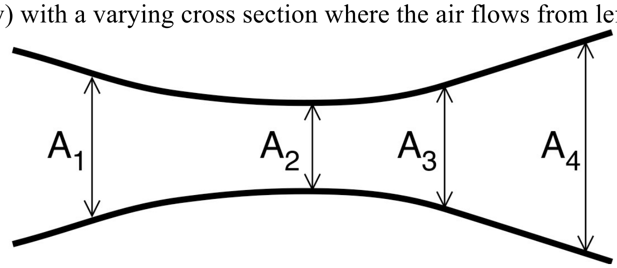
Problem 1-b

(i) Discuss how the static temperature, static pressure, total temperature, total pressure and flow velocity changes in:
- a supersonic flow without shock waves in a convergent channel
- a uniform supersonic flow turned into itself
- a uniform supersonic flow away from itself

(ii) Derive the relation between the total pressure p_0, static pressure p and the Mach number M for an isentropic flow, starting from the energy equation.

Problem 2-a

Consider a channel (see figure below) with a varying cross section where the air flows from left to right:



At the location of A_1 the velocity is $170.5\ m/s$, the static temperature is $T_1 = 289\ K$ and the static pressure is $p_1 = 10^5\ N/m^2$. The smallest cross section of the channel is A_2, for which $A_2/A_1 = 0.76$. Also it is given that: $A_4/A_1 = 0.785$.
(i) Compute the total pressure, Mach number and velocity at location A_4
(ii) What is the minimum value for A_2/A_1 so that the choked mass flow regime is reached?
(iii) For the minimum value of A_2/A_1 as computed in (ii) a normal shock wave is generated in A_3. Directly upstream of this shock, the Mach number is 1.2. For these conditions compute at location A_4: the total pressure, Mach number and velocity of the flow.

Problem 2-b

An aircraft is flying at an altitude of *10 km* with a speed of *2156 km/hour*. At that altitude the temperature is $-60\ ^oC$ and the pressure is $2.27 \times 10^4\ N/m^2$
(i) Compute the aircraft flight Mach number
(ii) Determine the pressure and the temperature experienced at the stagnation point on the nose of the aircraft

Appendix

Universal gas constant: $R_0 = 8314\ J/Kmol\ K$, specific heat of air: $C_p = 1004\ J/Kg\ K$