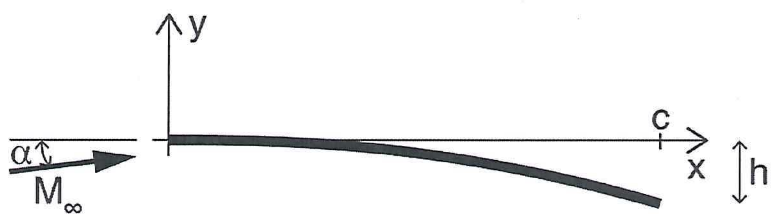
Delft University of Technology **DEPARTMENT OF AEROSPACE ENGINEERING**		
Course: Aerodynamics 2;	Code: AE2210	Course year: 2
Date: Wednesday 14 August 2013		Time: 14:00 – 17:00
On the top of <u>each</u> answer sheet write: initials, name, student number, sheet number/total number of sheets **This exam consists of 5 questions.**		

Problem 1

An airfoil is immersed in a supersonic flow at a free stream Mach number $M_\infty = 2.5$. The airfoil geometry is given by the equation $y = \frac{h}{c}\left[\cos\left(\frac{\pi x}{c}\right) - 1\right]$ with $\frac{h}{c} = 0.1$ and the angle of attack is $\alpha = 2^o$. You may use linear theory to answer this question.



i. What are the assumptions for which linear theory may be applied?

ii. Compute the pressure coefficient for the lower and upper side as function of x and plot them in a figure.

iii. Determine the lift and drag coefficient for this airfoil. Hint: $sin^2(x) = \frac{1}{2}(1 + \cos(2x))$

Problem 2

i. Consider a stationary oblique shock wave in a supersonic flow. Indicate what happens to the following variables when crossing the shock (stay constant, increase or decrease), justify you answer:
   a. Tangential Mach number
   b. Static enthalpy
   c. Critical Mach number (M*)

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 3

The SR71 aircraft is flying at an altitude of *22 km*, here the pressure is *4 kPa* and the temperature is *-55 °C*. A pitot tube positioned on the nose of the aircraft measures a pressure of *51.4 kPa*. What is the temperature in the stagnation point of the aircraft?

Problem 4

The Space Shuttle Main Engine is tested in a test stand at sea level ($p_{amb} = 1 \times 10^5\ Pa$). In the combustion chamber of the engine the total pressure is $p_o = 205 \times 10^5\ Pa$ and the total temperature is $T_o = 3315\ ^oC$. The exhaust diameter of the nozzle is $2.3\ m$ and the throat has a diameter of $0.26\ m$. 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. The thrust produced by a rocket engine is given as $F = \dot{m}V_{exit} + (p_{exit} - p_{amb})A_{exit}$. What is the thrust that is produced by the engine?

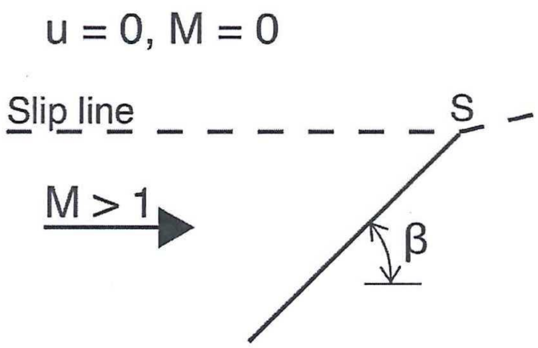
ii. Suddenly a turbopump fails and the pressure in the combustion chamber drops to $p_o = 30 \times 10^5\ Pa$ but the temperature stays the same. What is the thrust produced in this case?

A/A^*	M	A/A^*	M	A/A^*	M	A/A^*	M
2	0.3122	40	0.0148	2	2.0551	40	4.2394
4	0.1498	50	0.0118	4	2.6194	50	4.3958
6	0.0992	60	0.0099	6	2.9173	60	4.5245
8	0.0742	70	0.0085	8	3.1219	70	4.6340
10	0.0593	80	0.0074	10	3.2783	80	4.7294
20	0.0296	90	0.0066	20	3.7585	90	4.8140
30	0.0197	10	0.0059	30	4.0391	100	4.8900

*Mach – Area relation for $\gamma = 1.2$, please interpolate for intermediate values of M or A/A**

Problem 5

Consider a slip line delimiting a supersonic flow on the bottom ($M > 1$) with a stagnant region on the top ($u = 0,\ M = 0$). On the lower side there is an oblique shock wave that will interact with the slip line in point *s* (see figure below). Will the shock reflect from the slip line as a shock wave or as an expansion wave, justify your answer?



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