Aerodynamics C - Exam January 2005 Problems

1 Engine Intake of an Aircraft

1. Consider the air flow through the engine-intake of an aircraft flying at Mach 4 at an altitude of $10000m$ ($p = 2.65 \cdot 10^4 N/m^2$, $T = 223K$). The intake has a protruding wedge which deflects the flow by 30°. The flow is further decelerated to the subsonic regime with a normal shock wave that follows the first shock.
   (a) Draw a schematic of the intake geometry, including flow streamlines and shock waves.
   (b) Determine the value of the pressure and the temperature in the subsonic flow region where it is assumed that $M = 0.2$.
2. Determine the value of the specific heat ratio γ for Helium (He), Nitrogen (N_2) and Carbon Dioxide (CO_2) at ambient conditions. (Hint: $\gamma = (n+2)/n$ where n is the number of degrees of freedom of the molecular motion).

2 Designing a Supersonic Wind Tunnel

1. We need to design a supersonic wind tunnel reproducing at $M = 2.8$ free-stream flow. The cross section of the test section area is $A_{test} = 1m^2$. The air in the reservoir is at ambient temperature $T_0 = 280K$.
   (a) Determine the nozzle throat cross section $A_{t,1}$.
   (b) Determine the total pressure in the reservoir P_0 and the mass flow $\dot{m}$ through the nozzle in the hypothesis that the nozzle discharges the air directly in the ambient ($p_{amb} = 1atm$, $T_{amb} = 280K$) at supersonic conditions and without any shock.
   (c) A supersonic diffuser is placed behind the test section. Now the flow is decelerated through a normal shock wave at $M_{shock} = 2.8$. Determine the diffuser throat minimum cross section $A_{t,2}$, the minimum total pressure P_0 in the reservoir and the mass flow through the wind tunnel.
2. Consider air at a pressure of $0.3atm$. Calculate the values of the isothermal compressibility τ_T and isentropic compressibility τ_S expressing them in SI units.

3 Analyzing a Supersonic Airfoil

1. A semi convex airfoil is immersed in a uniform supersonic flow at Mach $M_\infty = 2.5$ with an incidence angle $\alpha = 3$ degrees, as is shown in figure 1.

   The upper surface has the following equation with respect to the body system of reference:

$$y = -\frac{4hx^2}{c^2} + \frac{4hx}{c}, \tag{0.1}$$

   where $h/c = 0.02$. The lower surface is flat. Determine:
   (a) The pressure coefficient distribution $C_p(x)$ on the bottom and top side of the airfoil.
   (b) The airfoil lift coefficient C_L.
   (c) The airfoil drag coefficient C_d.
2. Give the definition of the critical Mach number M_{cr} for an airfoil.

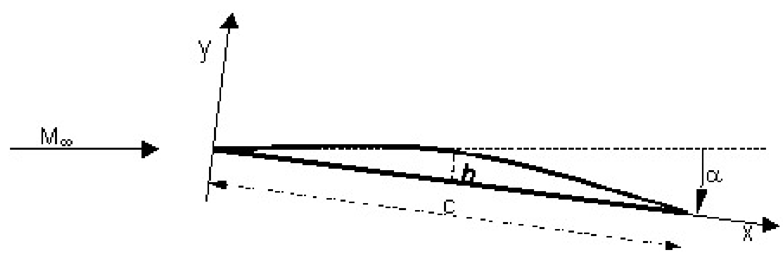

Figure 1: The airfoil corresponding to this question.