MAE 104 - SUMMER 2015
HOMEWORK 1

Due Tuesday 08-11-2015 in class

Guidelines:
Please turn in a *neat* homework that gives all the formulae that you have used as well as details that are required to understand your solution. Required plots should be generated using computer software such as Matplotlib or Excel. Remember to specify all the units of your results.

Problem 1:
During the design and develop of the McDonnell Douglas F/A-18 Hornet (see Figure 1), several wind tunnel tests were conducted. Amongst them, a 16% model was tested at low speed ($M_\infty \approx 0.08$), providing measurements for the lift coefficient, c_L, as a function of the angle of attack, α, (see Figure 2), the polar of the airplane (see Figure 3) and the coefficient of moment, $c_{M_{c/4}}$, with respect to the 25% mean chord as a function of α (see Figure 4).

Let's now consider the final version of the airplane, in horizontal flight with constant velocity

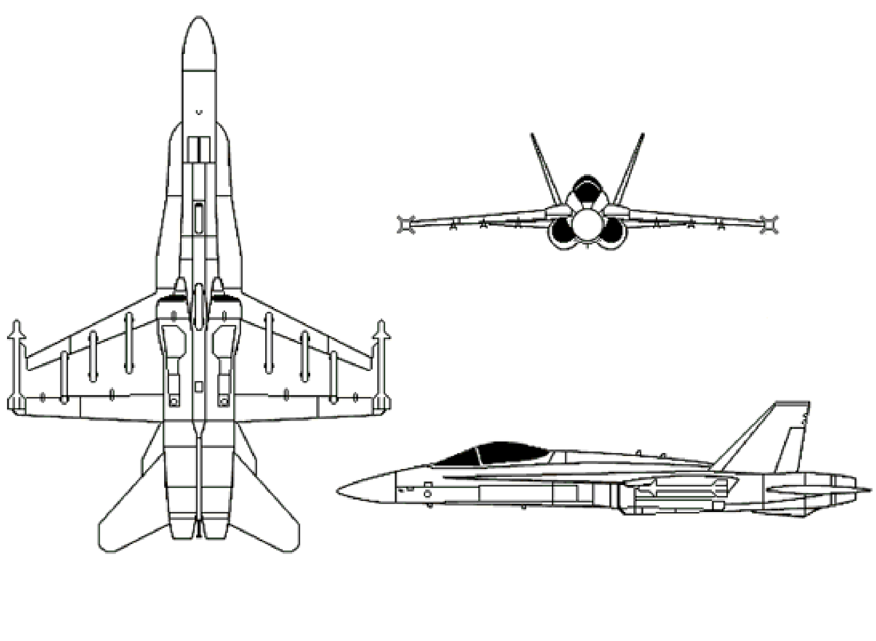

Figure 1: Schematic representation of the McDonnell Douglas F/A-18 Hornet [ref.].

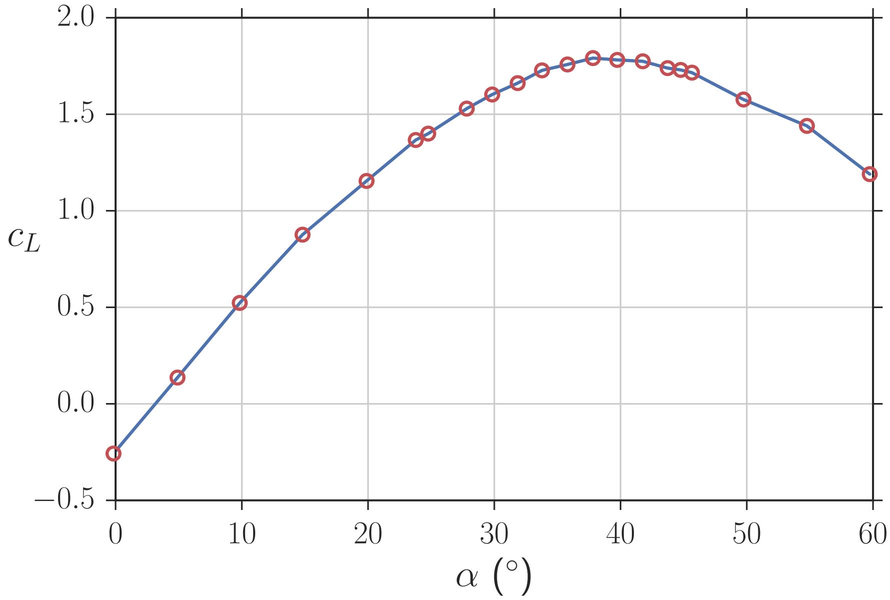

Figure 2: Lift coefficient of the airplane as a function of the angle of attack.

U_∞, and assume that its aerodynamic coefficients match those of the model. It presents a maximum take-off mass $M = 23,500$ kg, a wing area $S = 38$ m^2, a wingspan $b = 12.3$ m, a length $d = 17.1$ m and each of its two engines provide a maximum thrust $T = 79.2$ kN. The 25% mean chord is located at a distance from the front of 60% of the airplane length. Assume that the air density at the height of flight is $\rho = 1.23 kg/m^3$.

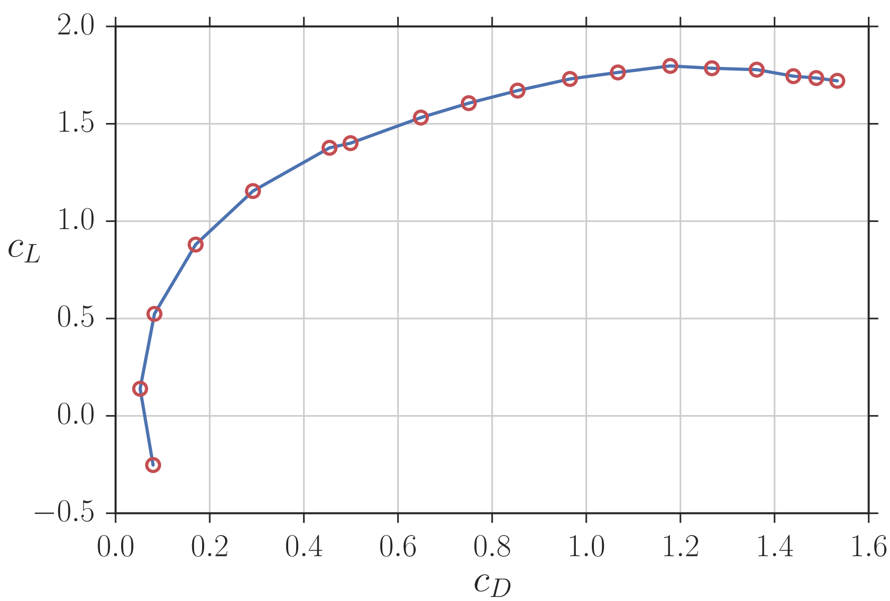

Figure 3: Polar of the airplane.

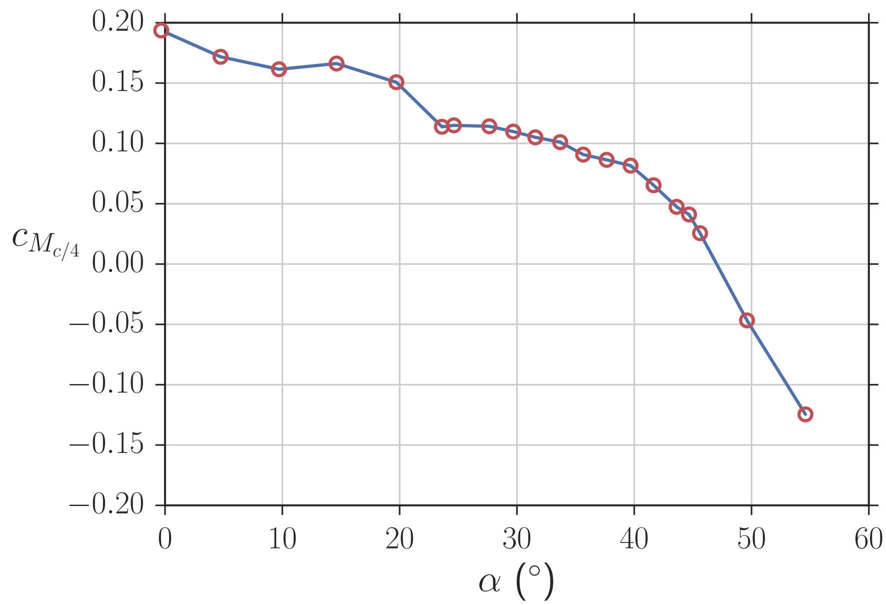

Figure 4: Coefficient of moment of the airplane with respect to the 25% mean chord.

1. Plot the drag coefficient, c_D, the aerodynamic efficiency, AE, and the center of pressure, x_{cp}, of the airplane as a function of the angle of attack.
2. Calculate the stalling velocity of the airplane. What are α, c_D, $c_{M_{c/4}}$ and AE at this flight condition?
3. Under the conditions described in the previous part, calculate the lift, drag, moment **with respect to the front of the airplane** M_0 and traction acting on the airplane.
4. Calculate the maximum velocity of the airplane. What are α, c_L, $c_{M_{c/4}}$ and AE at this flight condition?
5. Under the conditions described in the previous part, calculate the lift, drag, moment **with respect to the front of the airplane** and traction acting on the airplane.
6. What is the angle of attack that maximizes AE? Calculate the lift to drag ratio, velocity, lift L, drag D, moment **with respect to the front of the airplane** M_0 and thrust T for that angle of attack.
7. When the airplane is at a height $h = 4,000m$. both engines are turned off ($T = 0N$.) and the airplane is glided to the ground. Calculate the maximum distance that the airplane is able to glide at maximum lift to drag ratio.
8. Specify the units of all the previous results.

Bonus: Is there anything wrong with questions 4. and 5.?

Problem 2:
A thin airfoil of chord c is being tested in a wind tunnel. It is subjected to a free stream velocity U_∞ with an angle of attack α. For simplicity, we approximate its shape to a flat plate, as shown in Figure 5.

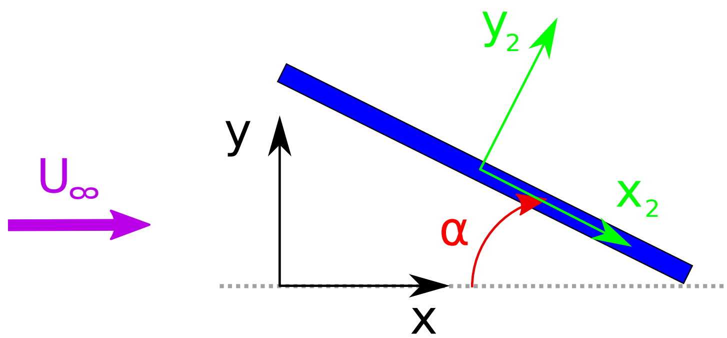

Figure 5: Approximation of the airfoil as a flat plate.

We place a coordinate system (x_2, y_2) on the center of the plate. In these coordinates, the coefficient of pressure in the upper and lower surfaces is measured to be

$$c_{p,u} = \frac{p_u - p_\infty}{\frac{1}{2}\rho U_\infty^2} = -2\alpha\sqrt{\frac{1-2x_2/c}{1+2x_2/c}} - \alpha^2 \cdot \frac{1-2x_2/c}{1+2x_2/c}$$

$$c_{p,l} = \frac{p_l - p_\infty}{\frac{1}{2}\rho U_\infty^2} = \quad 2\alpha\sqrt{\frac{1-2x_2/c}{1+2x_2/c}} - \alpha^2 \cdot \frac{1-2x_2/c}{1+2x_2/c}$$

as shown in Figure 6.

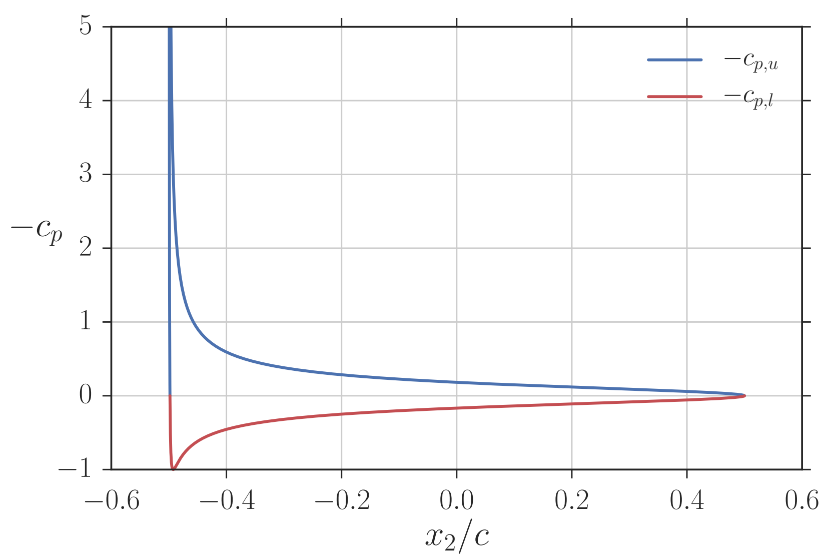

Figure 6: Pressure coefficient in the upper and lower surfaces of the airfoil.

Assuming that α is very small:

1. What is the relation between the lift L' and normal force N' on the airfoil? Justify why we can integrate along the reference system (x_2, y_2) instead of (x, y) to calculate L'.

2. Integrate c_p along the surface of the airfoil to calculate L'. **Hint:** use the coordinate system (x_2, y_2).

3. Calculate the coefficient of moment at the leading edge of the airfoil, $c_{m_{L.E.}}$.

4. Calculate the coefficient of moment at the $c/4$ point, $c_{m_{c/4}}$.

Note: You might consider the following integrals useful

$$\int_{\xi=-1}^{\xi=1} \sqrt{\frac{1-\xi}{1+\xi}} d\xi = \pi,$$

$$\int_{\xi=-1}^{\xi=1} \sqrt{\frac{1-\xi}{1+\xi}} \xi d\xi = -\frac{\pi}{2}.$$