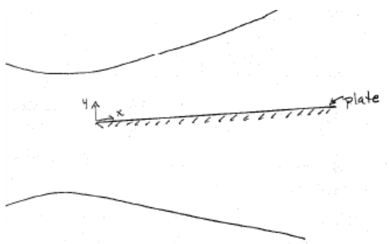
Problem Set #11 (Last!)

Due: Wednesday 12/04/2013

(2 problems; 20 points total)

1.[10 points] A flat plate of length l is placed inside a duct. By curving the walls of the duct, the pressure distribution on the flat plate can be set. Assume the walls of the duct are contoured in such a way that the potential flow over the plate give the following velocity on the surface of the flat plate: $u_e(x) = \dfrac{U_o}{x/l + 1}$ whre the plate starts at *x=0* and extends to positive values of *x*. The Reynolds number defined by $\mathrm{Re}_0 = \dfrac{U_o l}{\nu}$ has the value of 10,000.



a) Determine what portion of the plate have a favorable pressure gradient and what portion has an adverse pressure gradient.

b) Use Thwaites' method to find the non-dimensional momentum thickness, δ_2, as a function of x/l. Determine at x=0 (the beginning of the plate). As x becomes larges, the momentum thickness scales as $\delta_2 \propto x^n$, determine *n*. How does that compare with the Blasius (zero pressure gradient) boundary layer?

c) Calculate Thwaites' pressure gradient parameter, λ, as a function of *x/l*. Determine whether boundary layer separation occurs on the plate, and if so, where it is located (in terms of *x/l*).

2.[10 points] We will use XFLR5 program to do aerodynamic analysis for the NACA 2418 airfoil. XFLR5 is a GUI developed by Francesco Meschia (UC Berkeley) that uses the XFoil code by Mark Drela *et al* (MIT) for airfoil/wing analysis. The original version of XFoil is in Fortran and could be found here: http://web.mit.edu/drela/Public/web/xfoil/ You may download XFLR5 from: http://www.xflr5.com/xflr5.htm

In 333, we will simply use XFLR5 to analyze the 2D NACA 2418 airfoil. In this mode XFLR5 uses a 2D vortex panel method, similar to what we discussed in class. By default, XFLR5 considers viscous, boundary layer effects.

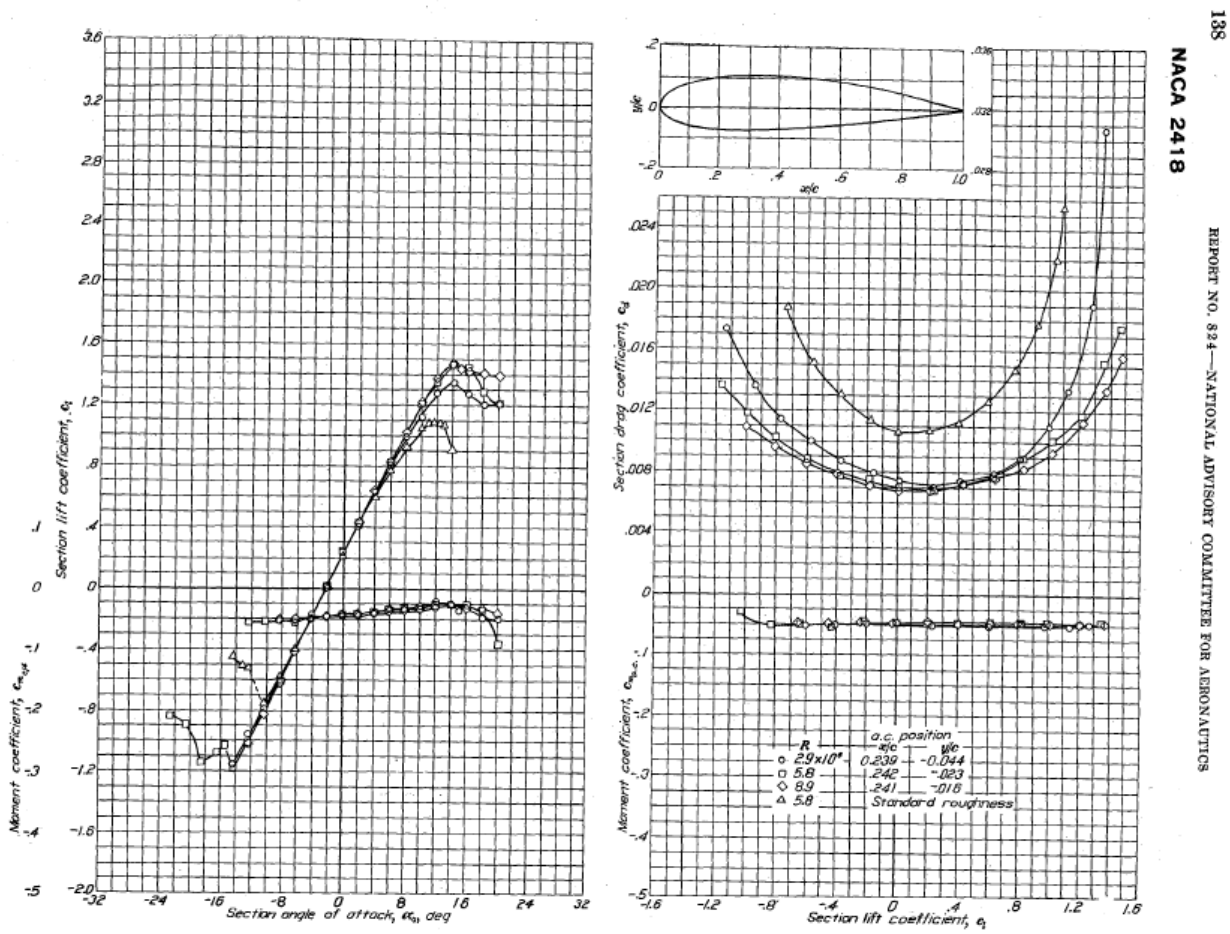
(a) Using the simple tutorial guide by Prof. Sullivan that could be found on piazza, run XFLR5 over a range of angles of attack from -8º to 12º and for chord Reynolds numbers of 2.9 x 10^6, 5.8 x 10^6, and 8.9 x 10^6. How does your solution change with the number of panels? Discuss the accuracy of your calculations.

(b) Plot C_l vs α and $C_{m,c/4}$ vs α for these three chord Reynolds numbers and compare with the classical experimental data for the NACA 2418 airfoil from Abbott and von Doenhoff ("Theory of Wing Sections," Dover Publications, 1949) which is given below. Comment on the similarities and/or differences in the theoretical and experimental results.

(c) For the lowest Reynolds number and two significantly different values of angle of attack, plot the variation of pressure coefficient C_p, displacement thickness D* and momentum thickness Theta and skin friction coefficient along the top and bottom airfoil surfaces. Comment on how these parameters change with the angle of attack and why.

(e) Plot the drag polar curves for different Reynolds numbers. Compare with experimental results from Abbott and Doenhoff, and note the transition points. Also find the drag (just at zero angle of attack) for lower Reynolds numbers and discuss the effect of Reynolds number on drag. Show any graph or table you deem necessary. Discuss your results including the accuracy of your calculations.

(d) How do lift and the center of pressure change when the angle of attack increases from -8º to 12º?



Aerodynamic characteristics of the NACA 2418 airfoil section, 24-inch chord.

Quote of the week:

"Knowledge and courage take turns at greatness."- Baltasar Gracian, 1601-1658.