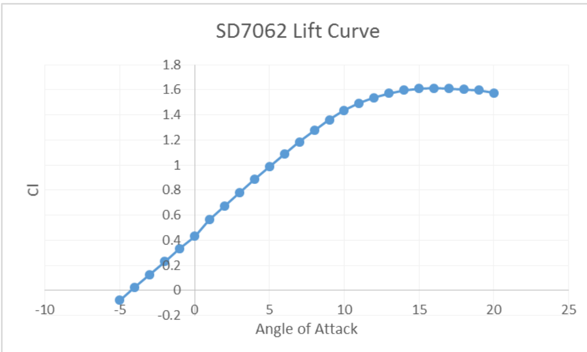
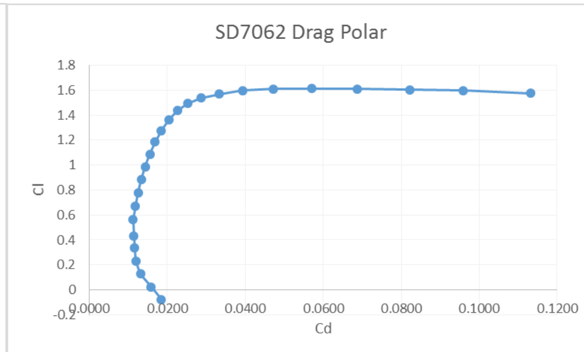
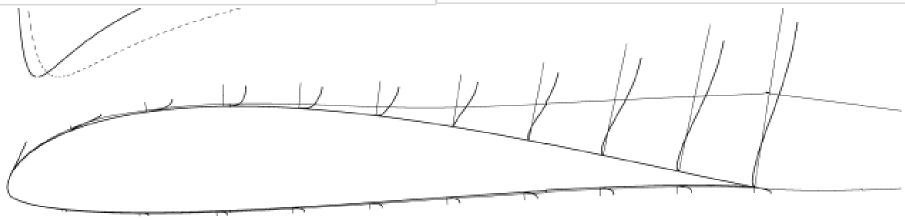
AEM 313 Problem Set #5

Solution

1. Using XFOIL analyze and plot for both the **NACA 4414** and **Selig Donovan SD7062** at Re=200000.
    - Lift Curves: C_l versus α
    - Drag polars: C_d versus C_l
    - Airfoils with boundary layer thickness and velocity profile at C_{lmax} (separate plots)

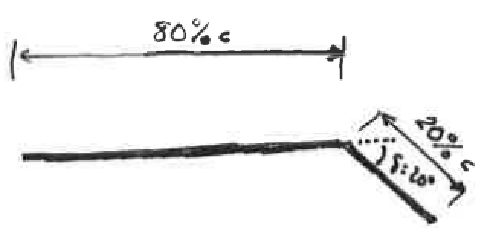






The 4414 is very similar (both are 14% thick, no a coincidence); however, the SD airfoil has a slight advantage near CLmax. CLmax for the SD is rather broad but is in the 14-16 degree range. Incidentally, the SD airfoil is WONDERFUL from a pilot perspective since the stall is soft and docile. The 4414 stall was slightly more abrupt. You should experiment with lowering and raising the Re to see how the BL and stall characteristics behave. XFOIL is an excellent intuition builder.

2. Use Thin Airfoil Theory to estimate the zero lift angle of attack and C_m at the quarter chord for a flat airfoil with a 20% flap deflected to 20 degrees.



First, solve for the flap's leading edge location in terms of the theta coordinates.

$$x = \frac{c}{2}(1-\cos(\theta)) \;\; \text{thus}$$

$$\theta_f = \arccos\left(1-2\frac{0.8}{1}\right) = 2.21 \text{ radians}$$

Next, from lesson 13, the slat-flap airfoil can be reduced to

$$\begin{aligned}
C_l &= 2\pi\alpha + 2\pi\delta_f\left(1-\frac{\theta_f}{\pi}+\frac{\sin(\theta_f)}{\pi}\right) \\
&= 2\pi\alpha + 2\pi\frac{20}{57.3}\left(1-\frac{2.21}{\pi}+\frac{\sin(2.21)}{\pi}\right) \\
&= 6.283\alpha + 1.2106
\end{aligned}$$

Solve for Cl=0 to give

$$\alpha_{zl} = -11.04^\circ$$

In this problem, we learned that a 20% flap deflected 20 degrees is approximately equal to 10 degrees AOA. This is a good value to remember.