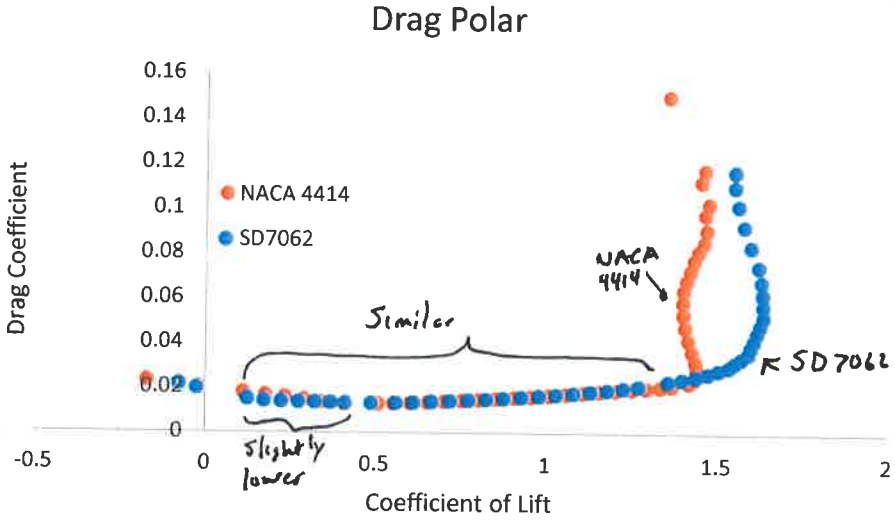
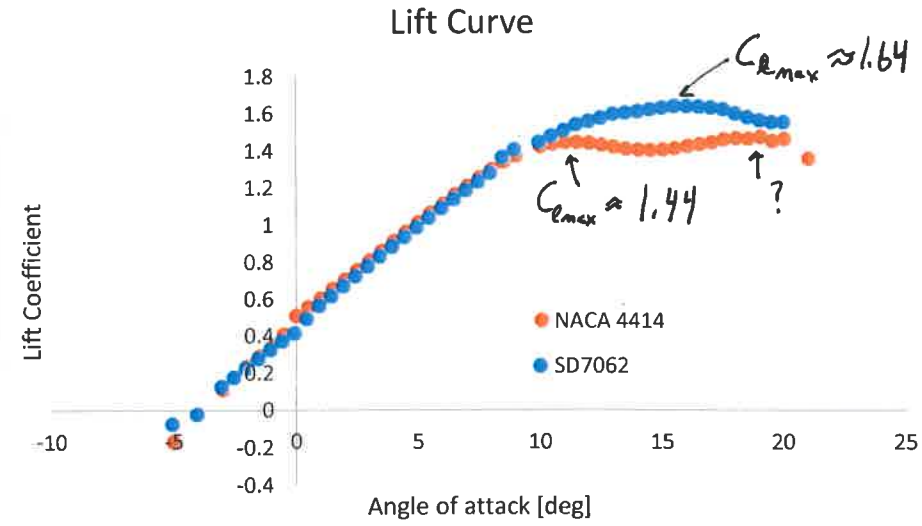
AEM 313
HW5

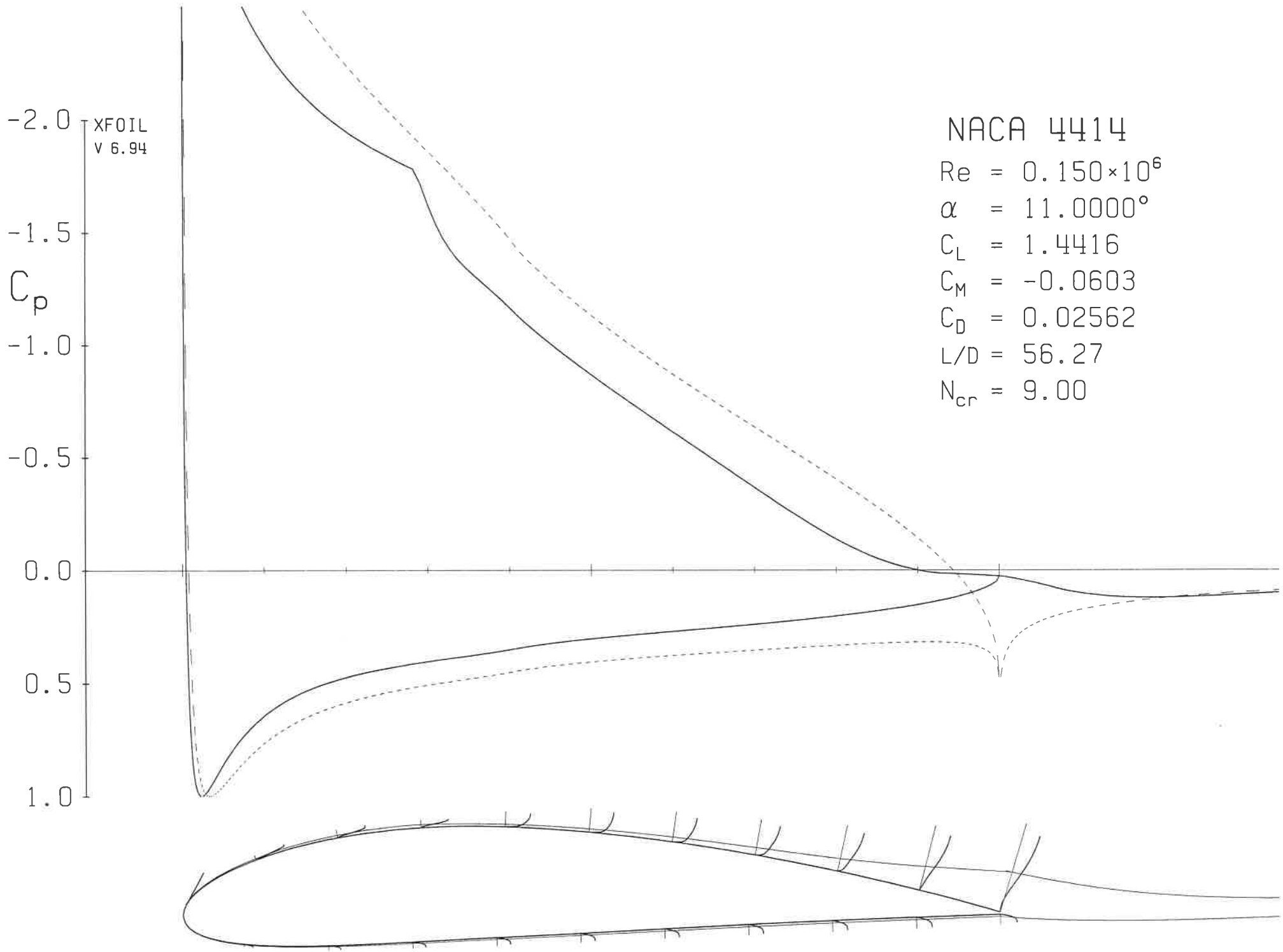
1) Compare the NACA 4414 and SD7062

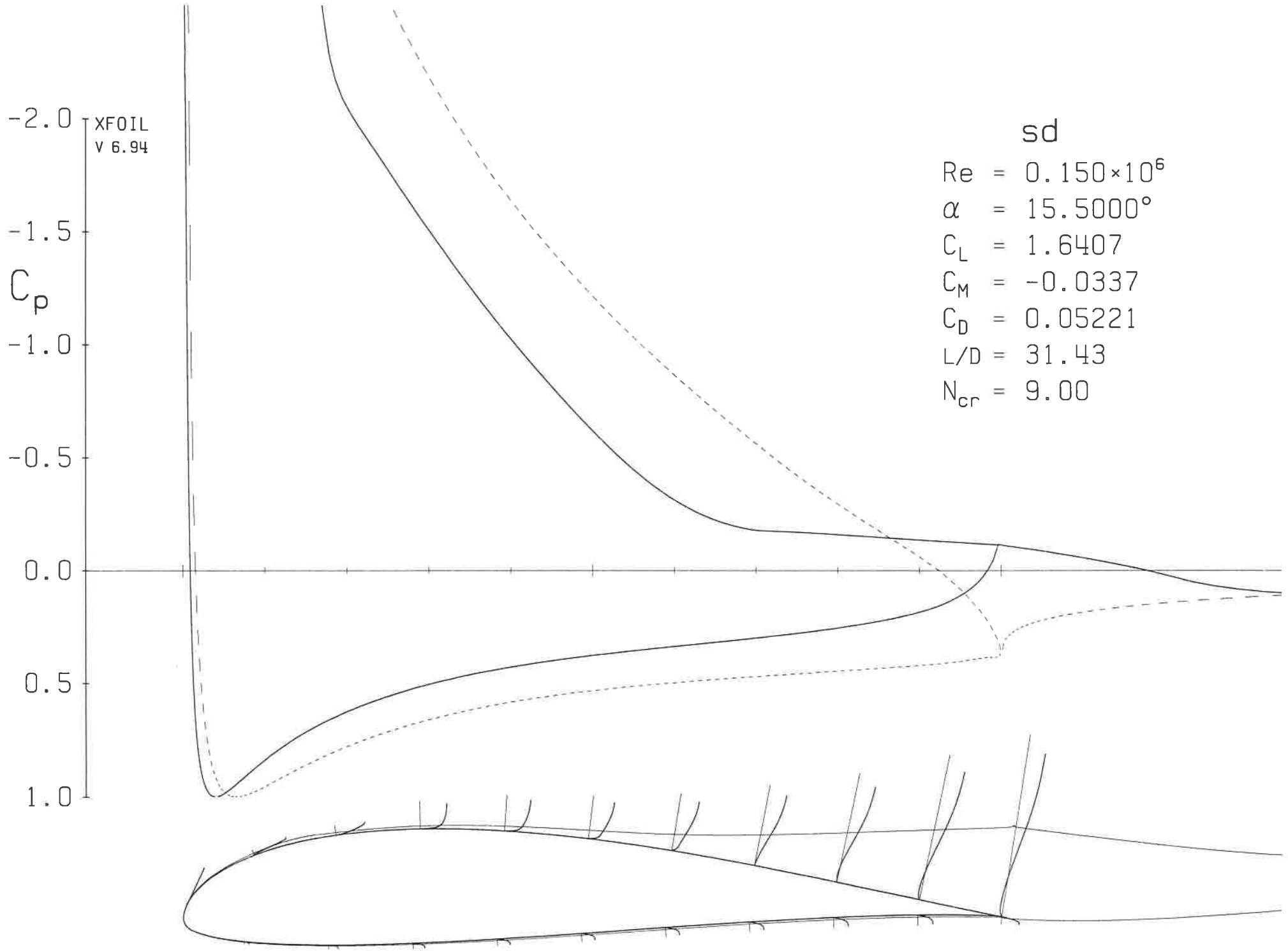


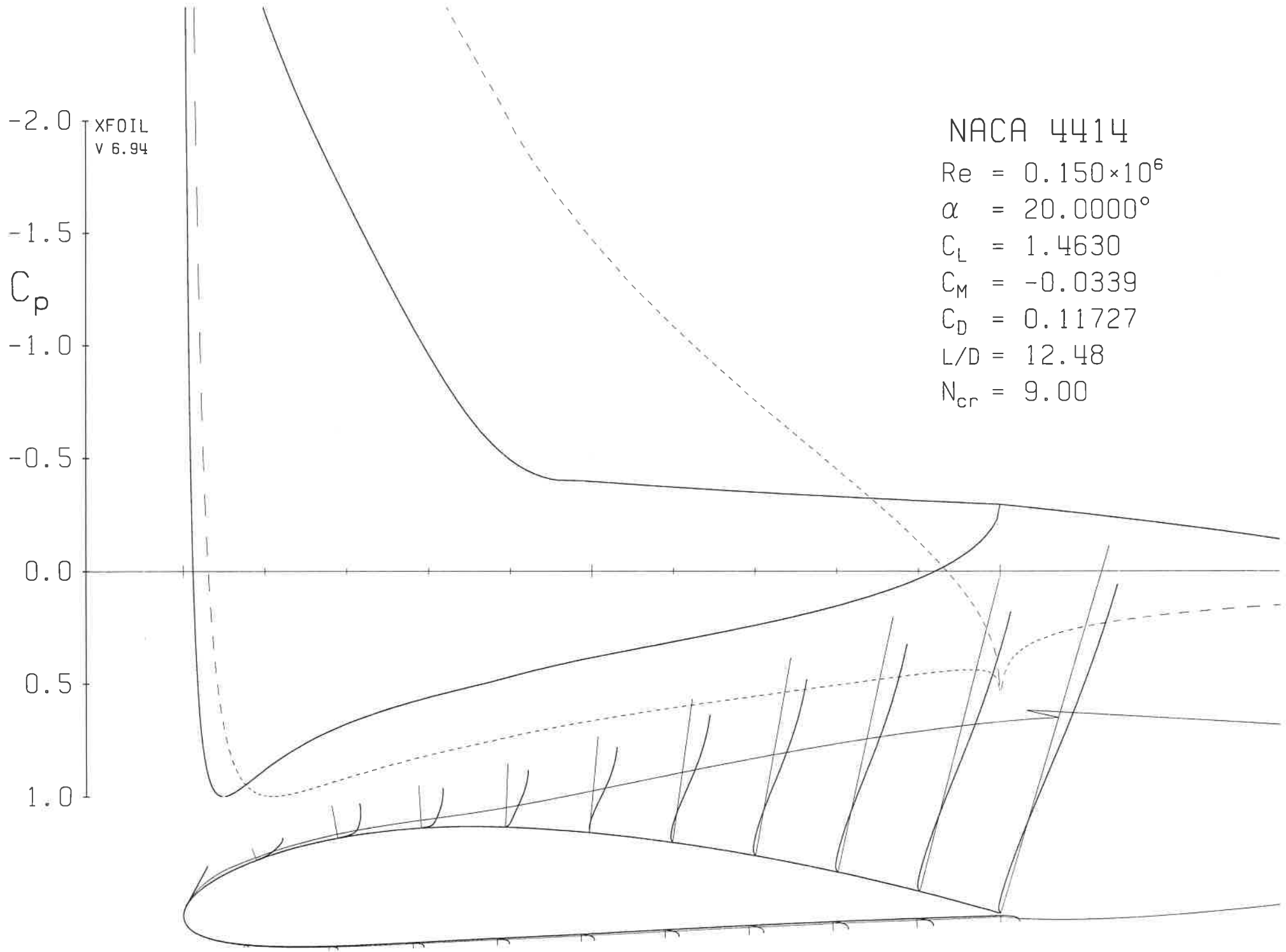


The SD7062 gives ≈ 14% more $C_{l_{max}}$ and a smoother roll off.

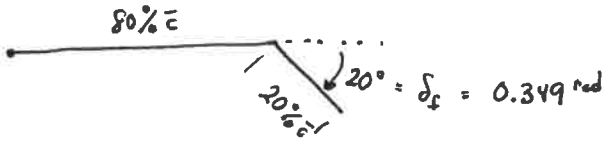
Overall, the SD7062 has an advantage in $C_{l_{max}}$, stall behavior, slightly lower C_d at low AOA, and significantly lower C_d vs C_l at higher C_l.


XFOIL
V 6.94
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
NACA 4414
Re = 0.150×10^6
α = 11.0000°
CL = 1.4416
CM = -0.0603
CD = 0.02562
L/D = 56.27
Ncr = 9.00


XFOIL
V 6.94
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
sd
Re = 0.150×10^6
α = 15.5000°
CL = 1.6407
CM = -0.0337
CD = 0.05221
L/D = 31.43
Ncr = 9.00


-2.0
-1.5
C_p
-1.0
-0.5
0.0
0.5
1.0
XFOIL
V 6.94
NACA 4414
Re = 0.150×10^6
α = 20.0000°
C_L = 1.4630
C_M = -0.0339
C_D = 0.11727
L/D = 12.48
N_{cr} = 9.00

2)



$20° = \delta_f = 0.349$ rad

Convert flap location $\frac{x}{c} = 80\%$ to cosine spacing,

$$x = \frac{c}{2}(1-\cos\theta) \quad \Rightarrow \quad \theta = \text{acos}\left(1 - 2\frac{x}{c}\right)$$

$$\theta_f = \text{acos}(1 - 2\cdot 0.8) = 2.21 \text{ rad}$$

From Lesson 13,

$$A_0 = \alpha - \frac{1}{\pi}\int_0^{\pi} \frac{dz}{dx}\, d\theta = \alpha - \frac{1}{\pi}\int_0^{\theta_f = 2.21} 0\, d\theta - \frac{1}{\pi}\int_{2.21}^{\pi} -\delta_f\, d\theta$$

$$= \alpha + \frac{1}{\pi}(\pi - 2.21)\underbrace{\delta_f}_{0.349 \text{ rad}} = \underline{0.103}$$

$$A_1 = \frac{2}{\pi}\int_0^{\pi} \frac{dz}{dx}\cos\theta\, d\theta = \frac{2}{\pi}\int_0^{2.21} 0\cdot\cos\theta\, d\theta + \frac{2}{\pi}\int_{2.21}^{\pi} -\delta_f \cos\theta\, d\theta = -\frac{2}{\pi}\delta_f \int_{2.21}^{\pi} \cos\theta\, d\theta$$

$$= 0.1778$$

$$A_2 = \frac{2}{\pi}\int_0^{\pi} \frac{dz}{dx}\cos 2\theta\, d\theta \quad \cdots\cdots > = -0.1067$$

Lift coeff.

$$C_\ell = 2\pi A_0 + \pi A_1 = 2\pi\alpha + 2\pi\cdot 0.103 + \pi\cdot 0.1778 = 2\pi\alpha + 1.21$$

$$\boxed{C_\ell = 2\pi\alpha + 1.21}$$

Moment

$$C_{m_{1/4c}} = -\frac{\pi}{4}(A_1 - A_2) = -\frac{\pi}{4}(0.1778 + 0.1067)$$

$$\boxed{C_{m_{1/4c}} = -0.2234}$$

Flaps retracted

$$\boxed{\begin{aligned} C_\ell &= 2\pi\alpha \\ C_{m_{1/4c}} &= 0 \end{aligned}}$$

3) plot A.C and C.P. for circular arc airfoil of 0%, 5%, 10% camber

- A.C is at 25% chord ....
- C.P (from lesson 13 notes) is at $X_{cp} = \frac{c}{4}\left(1 - \frac{4C_{mc/4}}{2\pi\alpha + C_{l_0}}\right)$

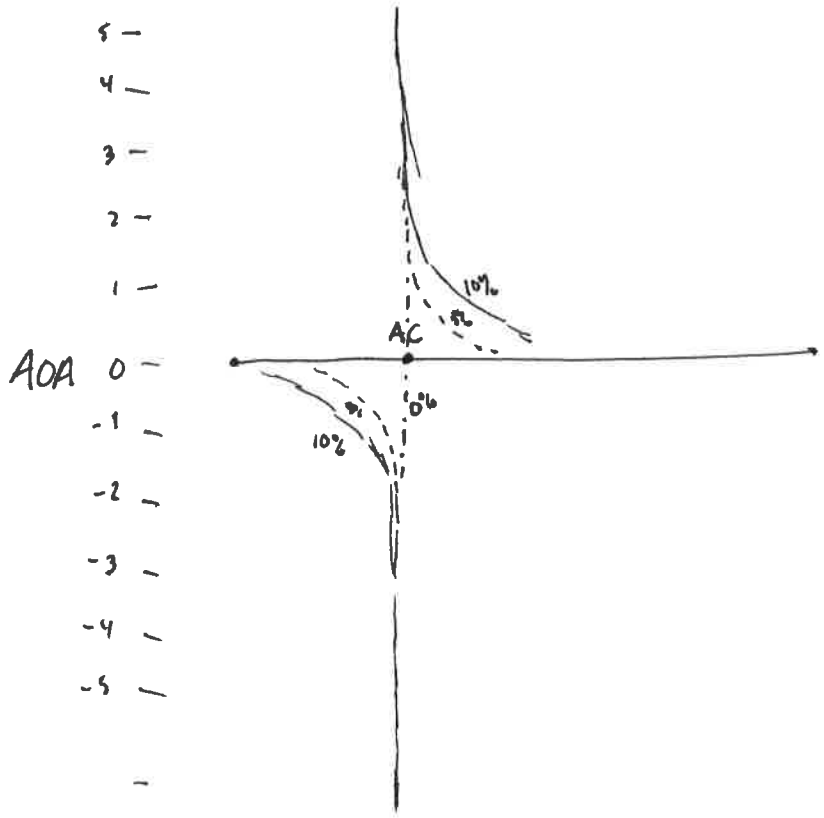
from lesson 13 notes

$$\begin{cases} A_0 = \alpha \\ A_1 = \frac{4f}{c} \\ A_2 = 0 \end{cases}$$

$$C_{m_{c/4}} = -\frac{\pi}{4}(A_1 - A_2) = -\frac{\pi}{4}\frac{4f}{c} = -\pi f/c$$

$$C_{l_0} = \pi A_1 = \pi \frac{4f}{c}$$

$$C_p = \frac{c}{4}\left(1 - \frac{4(-\pi f/c)}{2\pi\alpha + \pi 4f/c}\right)$$



4)

Lesson 13.

Flaps have Kutta Condition applied. Slats do not.