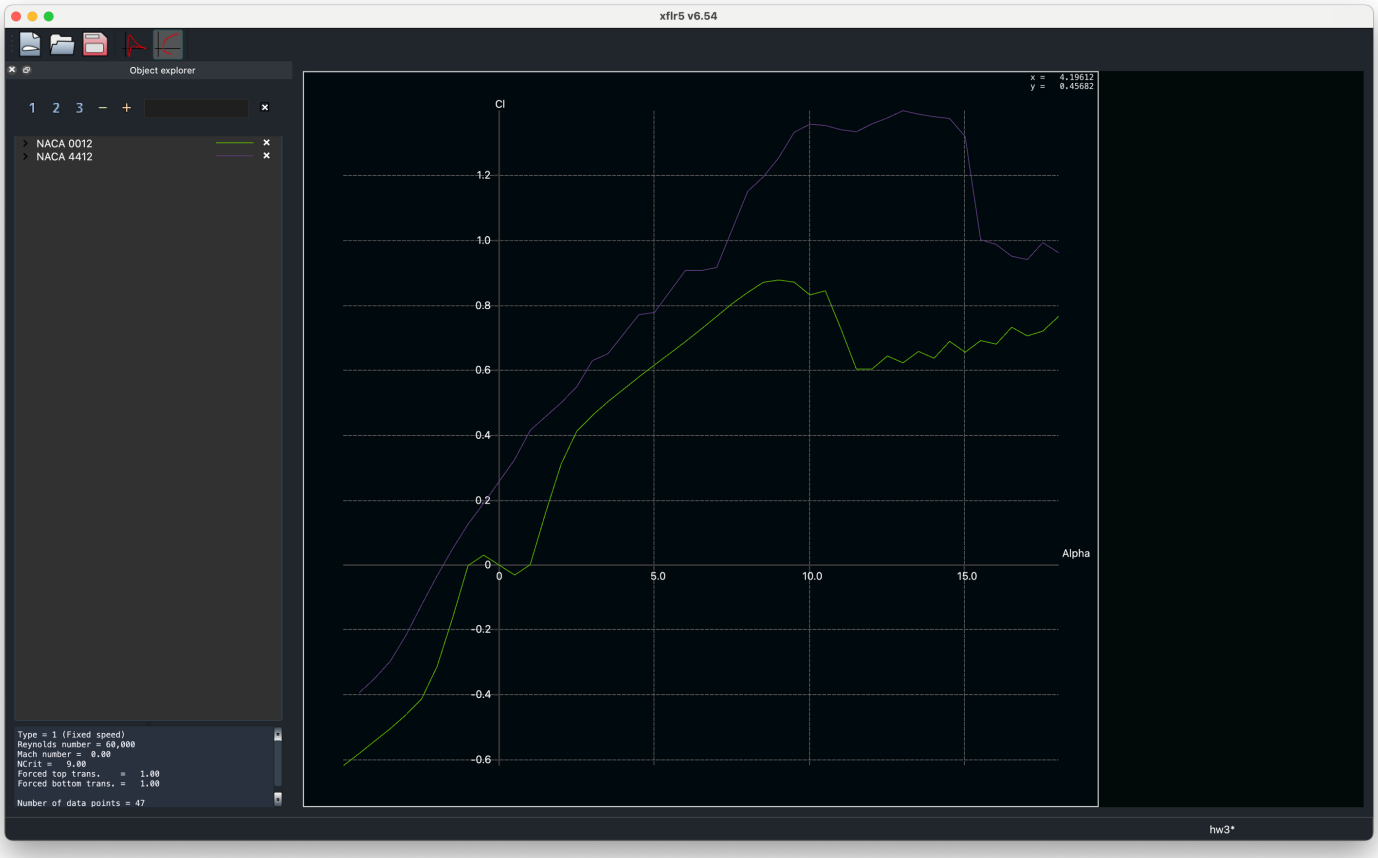
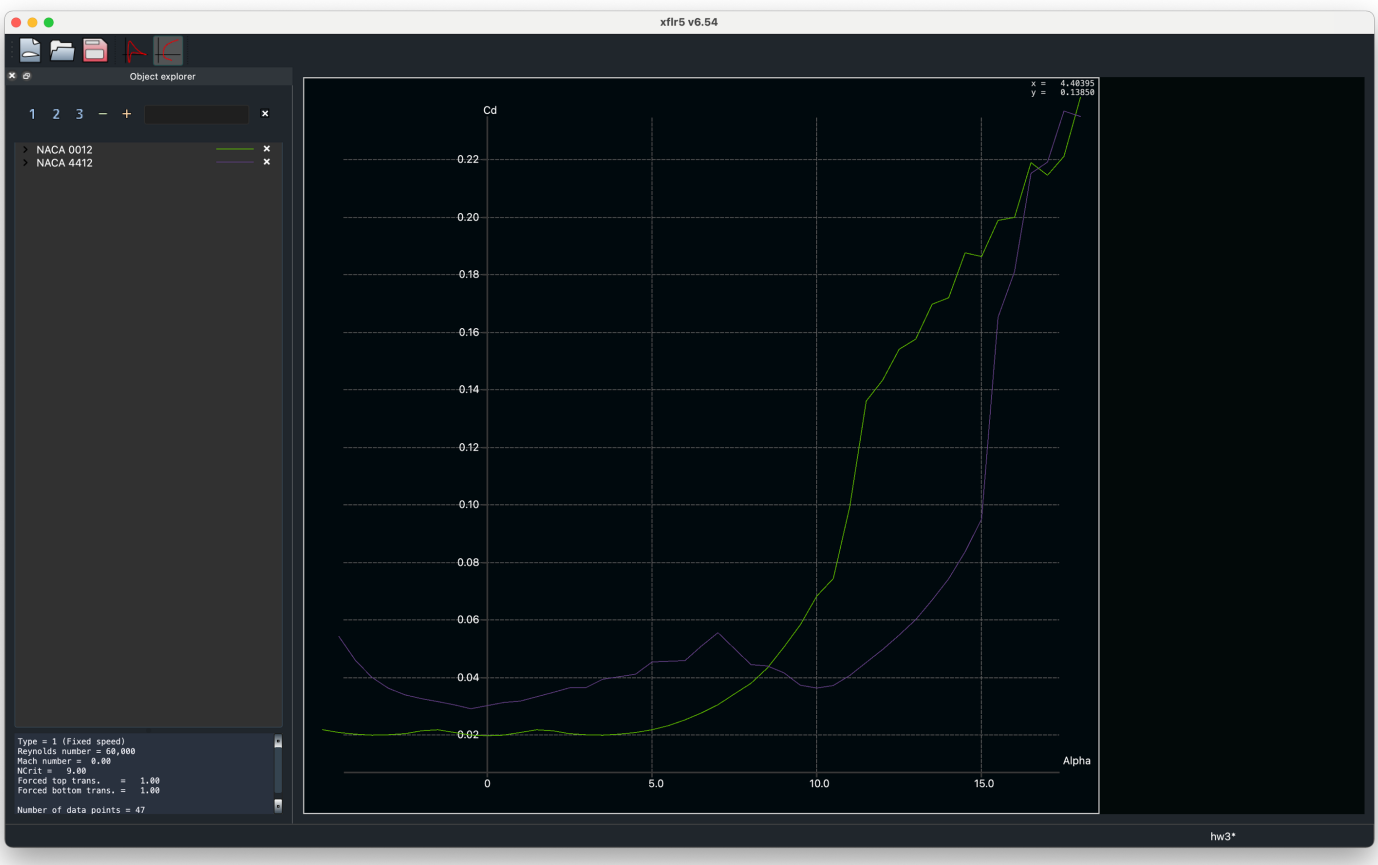
Problem 1

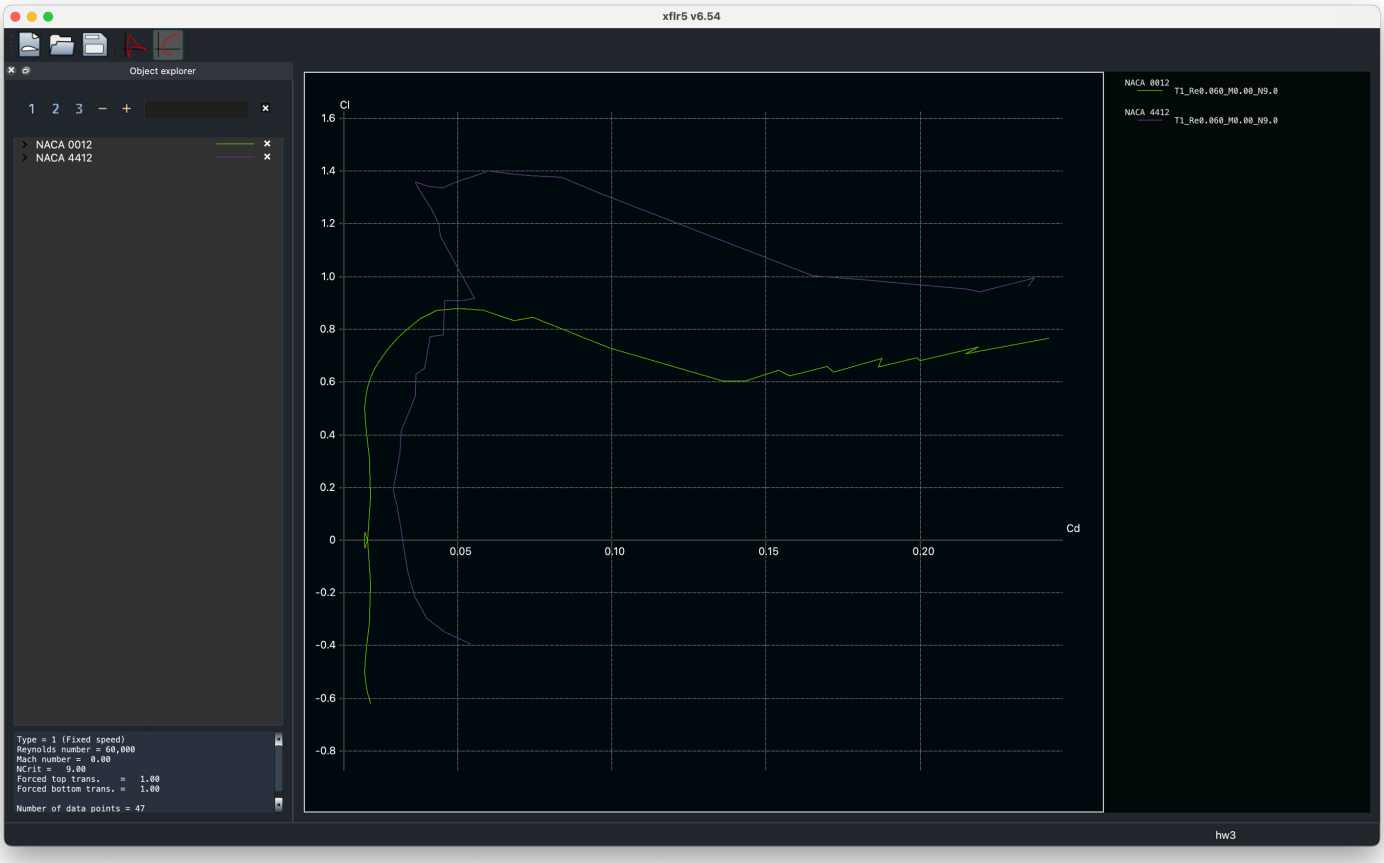
a)


xflr5 v6.54
Object explorer
1 2 3 − +
NACA 0012
NACA 4412
Type = 1 (Fixed speed)
Reynolds number = 60,000
Mach number = 0.00
NCrit = 9.00
Forced top trans. = 1.00
Forced bottom trans. = 1.00
Number of data points = 47
x = 4.19612
y = 0.45682
Cl
1.2
1.0
0.8
0.6
0.4
0.2
0
-0.2
-0.4
-0.6
0
5.0
10.0
15.0
Alpha
hw3*

b)


xflr5 v6.54
Object explorer
1 2 3 − +
NACA 0012
NACA 4412
Type = 1 (Fixed speed)
Reynolds number = 60,000
Mach number = 0.00
NCrit = 9.00
Forced top trans. = 1.00
Forced bottom trans. = 1.00
Number of data points = 47
x = 4.40395
y = 0.13850
Cd
0.22
0.20
0.18
0.16
0.14
0.12
0.10
0.08
0.06
0.04
0.02
0
5.0
10.0
15.0
Alpha
hw3*

c)

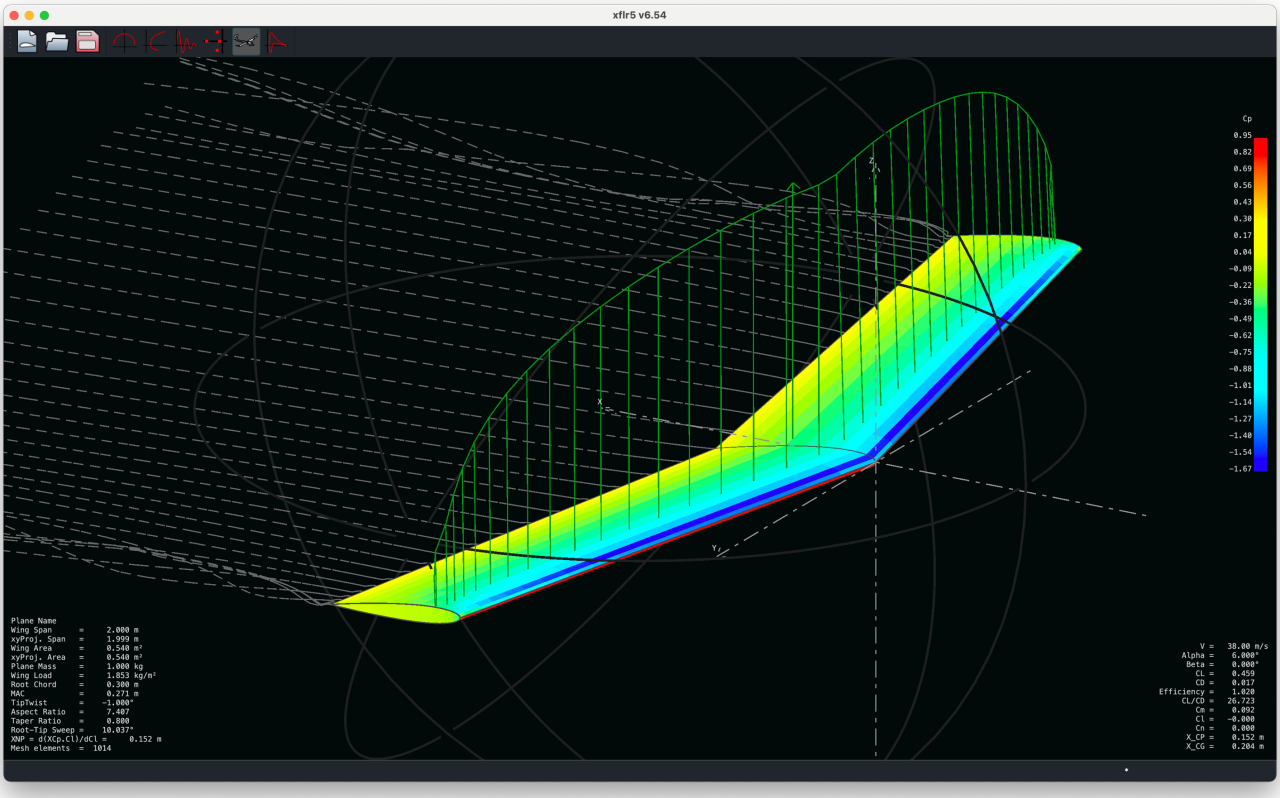


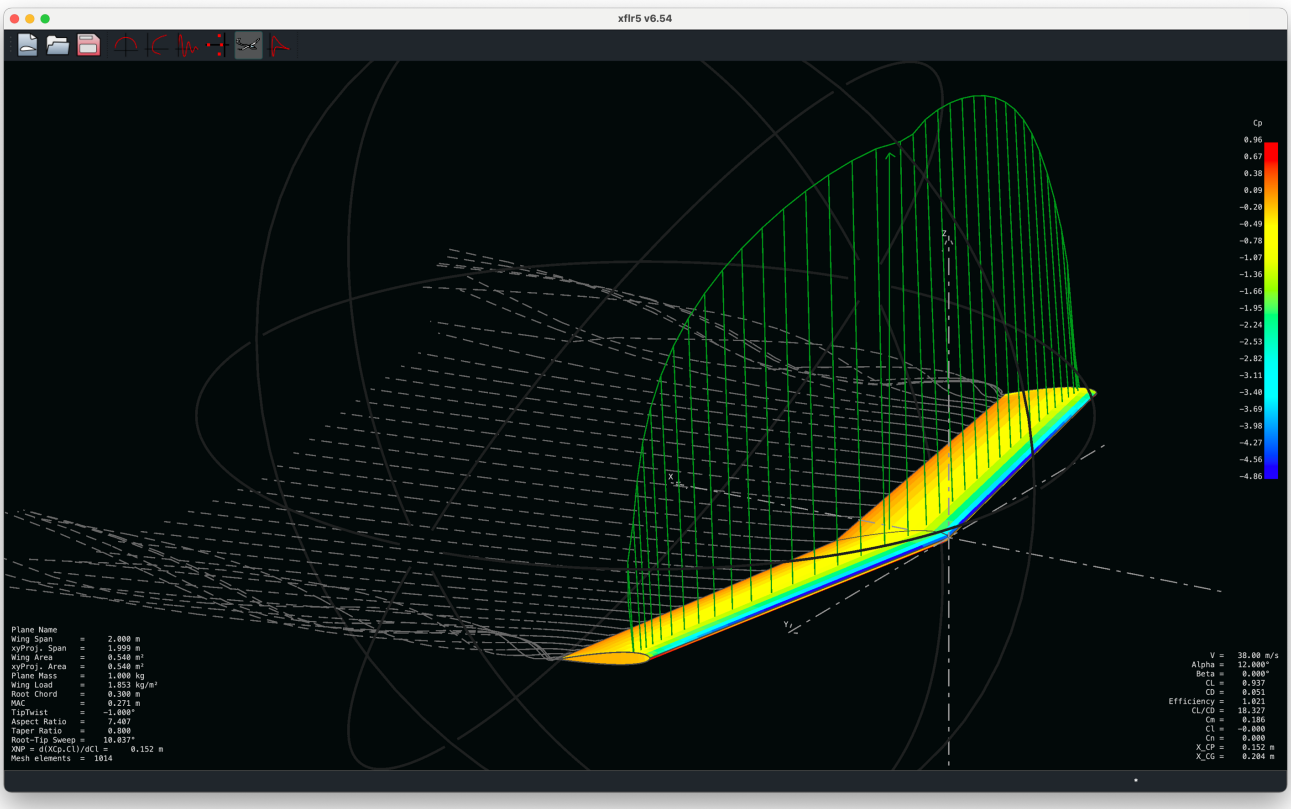
d) Comparing the two different airfoils, NACA 4412 has more camber than the symmetric NACA 0012. Looking at the Cl vs alpha graph, we can see how the NACA 0012 doesn't generate any lift at 0 degree angle of attack, while the NACA 4412 generates lift. We can all see the stall point delayed with the cambered airfoil and generates more lift with NACA 4412. If cambered is increased, stall is delayed and generates increased lift to a point before other drag affects take over.

Looking further into other properties, we can see that the Cl/Cd is significantly higher as well. This is likely due to the greater pressure differential between the upper and lower boundary of the airfoils. The increased camber generates a increase pressure differential, increasing the lift.

Problem 2

a) alpha = 6 deg -> CL = 0.4586, alpha = 12 deg -> CL = 0.9371
XCG = 0.204 m, XCP = 0.152 m





b) XNP = 0.152 m

Problem 3

a) The slope is negative. The trim angle is roughly an angle of attack of 2 deg.