A1. Write a code that uses the vortex-panel method outlined in section 1.6 to compute the lift coefficient, leading-edge pitching-moment coefficient, and quarter-chord pitching-moment coefficient at a specified angle of attack on an airfoil defined by a set of points that are read in from a text file. Use the points as the nodes in the algorithm.

For debugging purposes, the A matrix, B vector, and Gamma vector solutions for a NACA 2412 airfoil with 10 nodes at zero angle of attack can be viewed here. Results for this example should give the following:

$$\tilde{C}_L = 0.239837889916$$

$$\tilde{C}_{m_{LE}} = -0.111898723302$$

$$\tilde{C}_{m_{c/4}} = -0.0519392508233$$

A1 ANSWER: See the attached .py file to see my code!

A2. Use the code developed for A1 to fill out the following table for the NACA 2412 airfoil with 200 nodes to 5 decimal places. The input to the code should include the following:

```
{
    "geometry" : "2412_200.txt",
    "freestream_velocity" : 10.0,
    "alpha[deg]" : 0.0
}
```

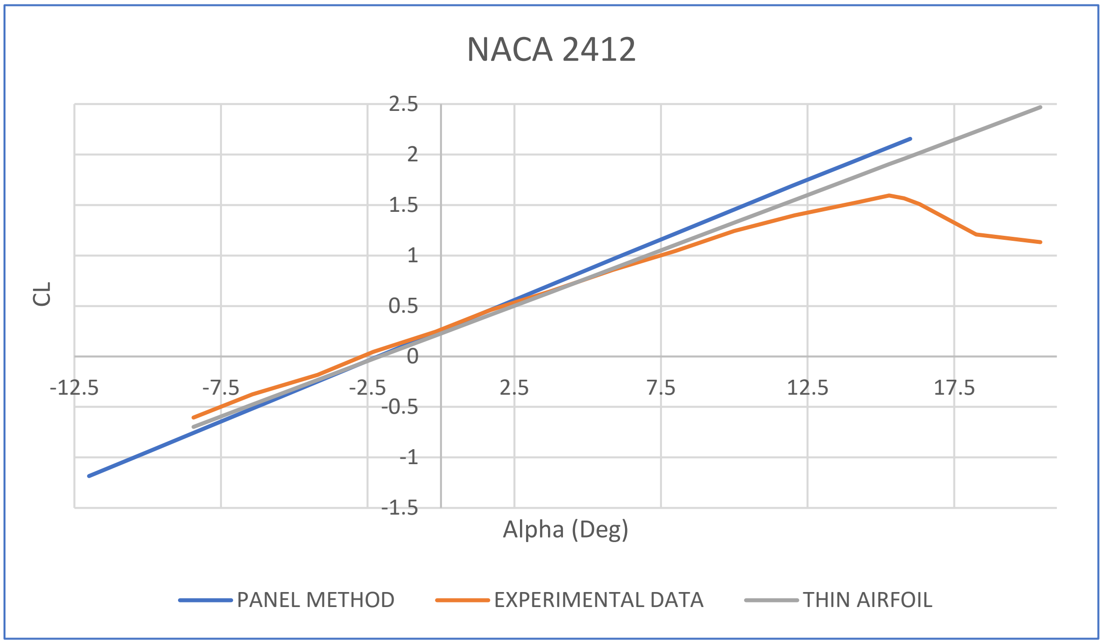
A2 ANSWER:

Table 1: NACA 2412 (200 pts, panel code)

α (deg)	$\widetilde{C_L}$	$\tilde{C}_{mle}$	$\tilde{C}_{mc/4}$
-12	-1.18434	0.249132	-0.040484
-10	-0.945636	0.190142	-0.0426751
-8	-0.705775	0.129729	-0.0449979
-6	-0.465055	0.0681856	-0.0474411
-4	-0.223767	0.00581275	-0.0499928
-2	0.0177923	-0.057086	-0.0526406
0	0.25933	-0.120204	-0.0553715
2	0.500553	-0.183234	-0.0581723
4	0.741165	-0.245869	-0.0610293
6	0.980874	-0.307804	-0.0639286
8	1.21939	-0.368736	-0.066856
10	1.45642	-0.42837	-0.0697974
12	1.69167	-0.486414	-0.0727383
14	1.92486	-0.542586	-0.0756644
16	2.15571	-0.596613	-0.0785616

A3. Plot your lift-coefficient solution from A2 in comparison to predictions from thin-airfoil theory and experimental data. The zero-lift angle of attack from thin-airfoil theory is −2.077 degrees. Experimental data can be obtained from these tables. Use the data at a Reynolds number of about 3 million. What do you notice about the results?

A3 ANSWER:



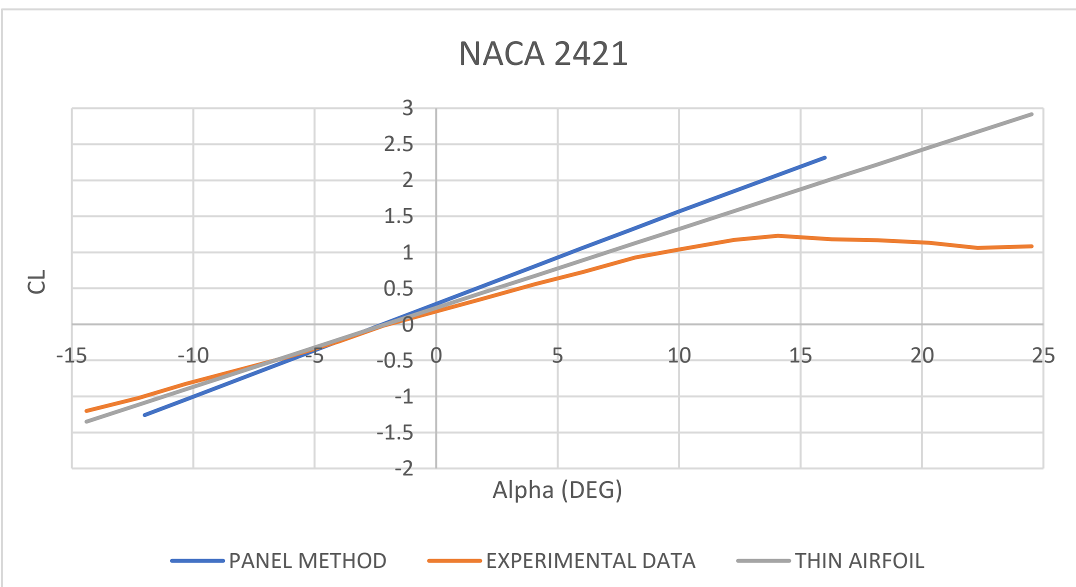
From these results, I notice that the Results from thin airfoil theory more closely reflect the experimental data than the results from the vortex panel code. This makes sense because thin airfoil theory neglects both thickness and viscosity effects. Increased thickness tends to create an increase in lift, while increased viscosity causes a decrease in lift. On the other hand, the vortex panel code takes thickness into account, but still neglects viscosity. Because of this, for airfoils that can reasonably be considered “thin”, thin airfoil theory should predict lift over a range of angles of attack more precisely than the vortex panel method. This is apparent in the data we collected here.

A4. Use your airfoil code to fill out the table shown in A2 for a NACA 2421 airfoil with 200 nodes. Plot your lift-coefficient results in comparison to predictions from thin-airfoil theory and experimental data. The zero-lift angle of attack from thin-airfoil theory is −2.077 degrees. Experimental data can be obtained from these tables. Use the data at a Reynolds number of about 3 million. What do you notice about the results?

A4 ANSWER:

Table 2: NACA 2421 (200 pts, panel code)

α (deg)	$\widetilde{C_L}$	$\tilde{C}_{mle}$	$\tilde{C}_{mc/4}$
-12	-1.25946	0.284108	-0.0238772
-10	-1.00407	0.218402	-0.0288011
-8	-0.747446	0.151076	-0.0339666
-6	-0.489916	0.08246	-0.0393483
-4	-0.231789	0.012886	-0.0449201
-2	0.0266206	-0.05731	-0.0506548
0	0.284997	-0.12777	-0.0565245
2	0.543027	-0.19818	-0.0625005
4	0.800395	-0.26817	-0.0685539
6	1.05679	-0.33741	-0.074655
8	1.31189	-0.40556	-0.0807742
10	1.5654	-0.47229	-0.0868816
12	1.817	-0.53727	-0.0929476
14	2.06639	-0.60019	-0.0989424
16	2.31326	-0.66075	-0.104837



As I mentioned in question A3, the thin airfoil results more closely resemble the results from the experimental data. This time there's a small difference, however. It's a little hard to see at first glance, but the line using thin airfoil theory for the NACA 2421 is a bit farther away on average from the experimental data than the thin airfoil theory line for the NACA 2412. This is to be expected because the 21 on NACA 2421 is indicative of a max thickness of 21% of the chord rather than a max thickness of 12%

of the chord in the NACA 2412. The thicker the airfoil, the less reliable thin airfoil theory is for predicting its lift.

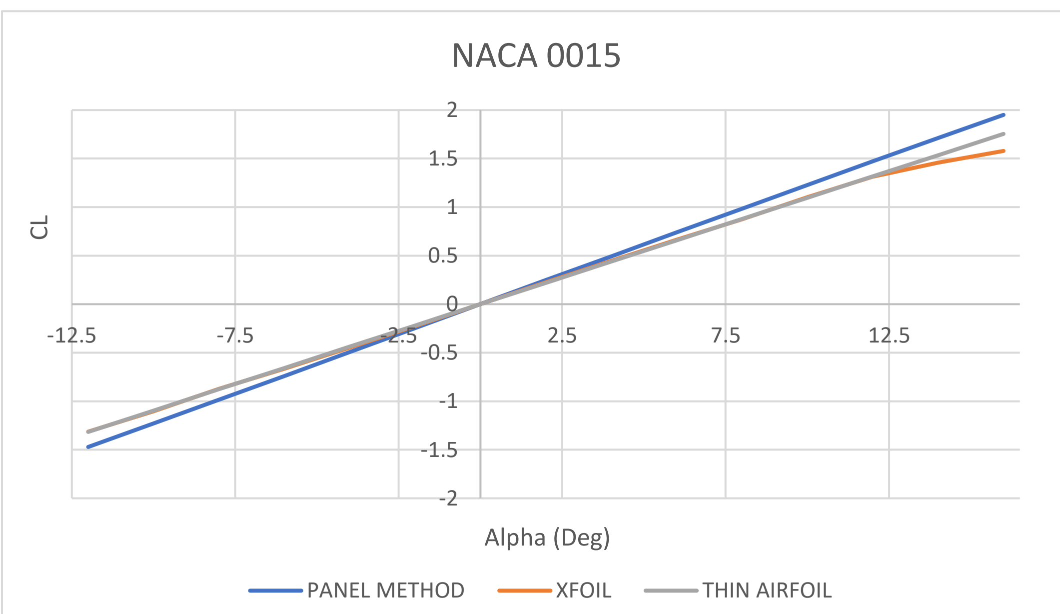
A5. Use your airfoil code to fill out the table shown in A2 for a NACA 0015 airfoil with 200 nodes. Use XFOIL to fill out the same table for the same airfoil at a Reynolds number of 3 million. Plot the lift-coefficient results from your panel code, XFOIL, and thin-airfoil theory. What do you notice about the results?

Table 3: NACA 2421 (200 pts, xfoil)

α (deg)	$\widetilde{C_L}$	$\tilde{C}_{mle}$	$\tilde{C}_{mc/4}$
-12	-1.3138	0.317966	-0.004
-10	-1.102	0.267791	-0.004
-8	-0.8738	0.210637	-0.006
-6	-0.6649	0.163511	-0.002
-4	-0.4474	0.11069	-0.001
-2	-0.2253	0.056342	0
0	0	0	0
2	0.2253	-0.05634	0
4	0.4474	-0.11069	0.001
6	0.6649	-0.16351	0.002
8	0.8738	-0.21064	0.006
10	1.1021	-0.26782	0.004
12	1.3138	-0.31797	0.004
14	1.4595	-0.33902	0.016
16	1.5797	-0.35307	0.028

Table 4: NACA 0015 (200 pts, panel code)

α (deg)	$\widetilde{C_L}$	$\tilde{C}_{mle}$	$\tilde{C}_{mc/4}$
-12	-1.47119	0.380935	0.021176
-10	-1.22874	0.320324	0.017807
-8	-0.98479	0.258152	0.014351
-6	-0.73965	0.194723	0.010825
-4	-0.4936	0.130345	0.007246
-2	-0.24695	0.065332	0.003632
0	4.09E-16	-2.97E-16	-1.95E-16
2	0.246949	-0.06533	-0.00363
4	0.493598	-0.13035	-0.00725
6	0.739645	-0.19472	-0.01082
8	0.984791	-0.25815	-0.01435
10	1.22874	-0.32032	-0.01781
12	1.47119	-0.38094	-0.02118
14	1.71184	-0.43969	-0.02444
16	1.95041	-0.4963	-0.02759



Something I notice about the results for this section, is that the XFOIL results very closely match the results of thin airfoil theory up until the angle of attack is around 12.5 degrees. Because I used a Reynolds number of 3 million, it appears that the inviscid assumption inherent in both the panel method and thin airfoil theory is appropriate here. I also noticed that all three lines intersect the y-axis at alpha=0. This makes sense. At zero angle of attack, a non-cambered airfoil like the NACA 0015 should not produce any lift.