Airfoil Analysis

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October 9, 2024

A11

The code was tested using the provided JSON input, where a single streamline spiraled outward from the vortex element. The resulting flow field displayed distinct interactions between the vortex and other flow elements, illustrating the dynamic nature of the combined potential flow elements.

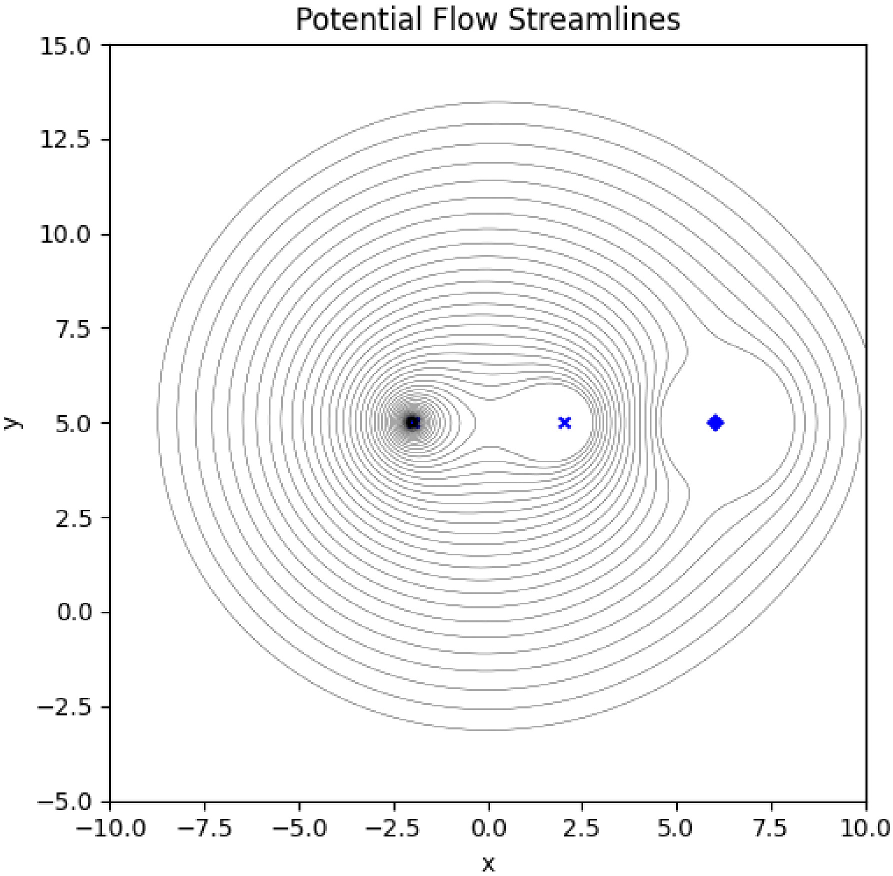


Figure 1: Potential Flow for the Provided Elements

A13: Table of α, C_L, C_{mle}, and $C_{m(c/4)}$ for a NACA 2412 Airfoil

α (deg)	C_L	C_{mle}	$C_{m(c/4)}$
-12	-1.18549	2.95898	2.66261
-10	-0.94631	2.50961	2.27303
-8	-0.70598	2.03142	1.85492
-6	-0.46478	1.52673	1.41053
-4	-0.22302	0.99800	0.94225
-2	0.01901	0.44782	0.45257
0	0.26102	-0.12115	-0.05589
2	0.50271	-0.70612	-0.58044
4	0.74379	-1.30425	-1.11830
6	0.98396	-1.91262	-1.66663
8	1.22293	-2.52828	-2.22255
10	1.46041	-3.14821	-2.78311
12	1.69612	-3.76941	-3.34538
14	1.92975	-4.38883	-3.90640
16	2.16104	-5.00348	-4.46322

Table 1: Comparison of C_L, C_{mle}, and $C_m(c/4)$ at different angles of attack

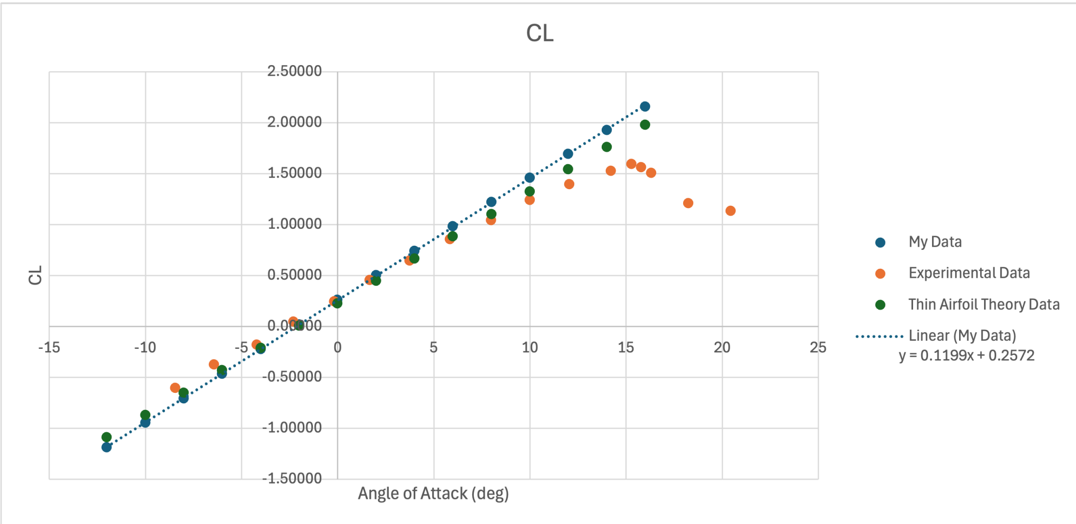


Figure 2: Comparison of Lift Coefficient for NACA 2412 Airfoil (A13)

What do you notice about the results?
My data shows the highest slope compared to both thin airfoil theory and experimental data. The difference between the inviscid solution and experimental data becomes more pronounced at higher angles of attack.

A14: NACA 2421 Airfoil Lift Coefficient Comparison

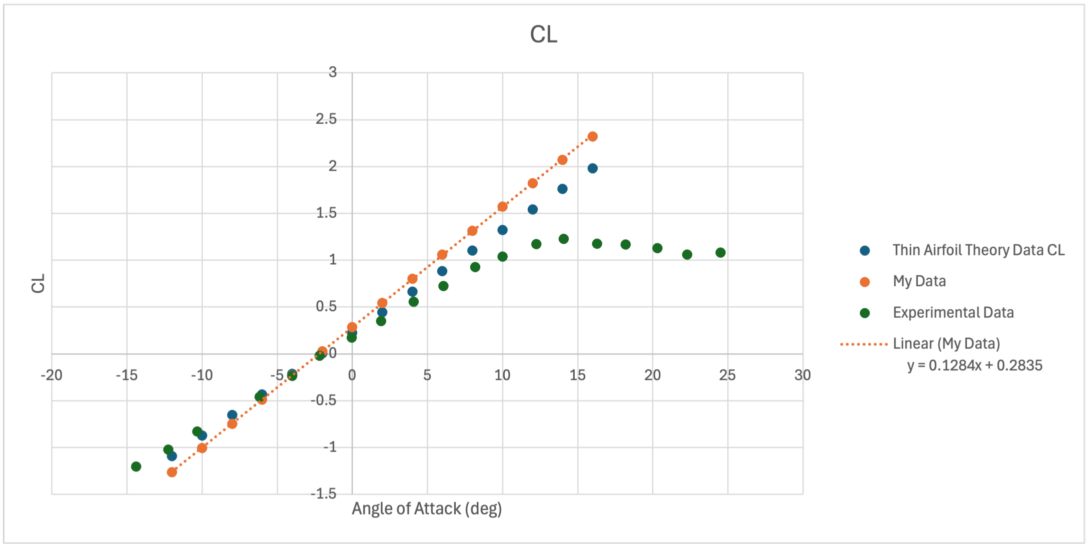


Figure 3: Comparison of Lift Coefficient for NACA 2421 Airfoil (A14)

What do you notice about the results?
As with the NACA 2412 airfoil, my data has the steepest slope compared to both thin airfoil theory and experimental data, with a larger difference at higher angles of attack. The difference occurs at a lower angle of attack compared to the NACA 2412 airfoil.

A15: NACA 0015 Airfoil Lift Coefficient Comparison

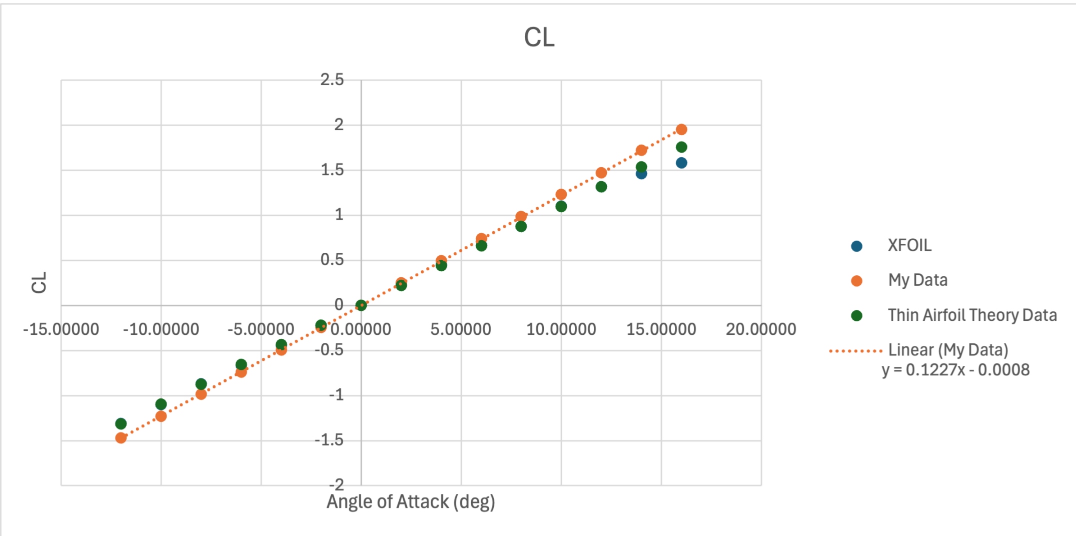
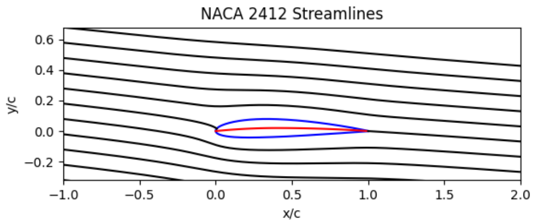


Figure 4: Comparison of Lift Coefficient for NACA 0015 Airfoil (A15)

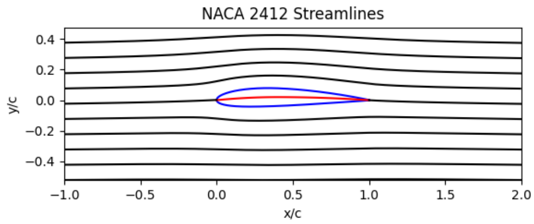
What do you notice about the results?
For the NACA 0015 airfoil, my data still shows the highest slope, while thin airfoil theory and the XFOIL data predict lower lift coefficients, especially at higher angles of attack. XFOIL has additional calculations that demonstrate a more realistic lift slope. The difference between my data and the other data is much smaller than the other two airfoils.

A17: Streamlines for NACA 2412 Airfoil

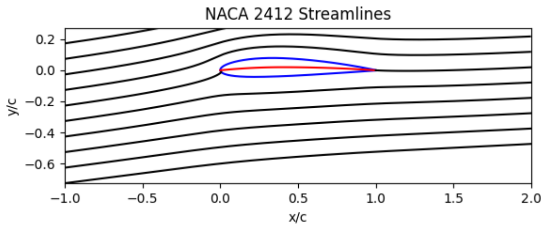
Use the code developed for problem A16 to plot the streamlines around the NACA 2412 airfoil at angles of attack of -5, 0, and 5 degrees.



(a) NACA 2412 at -5 degrees



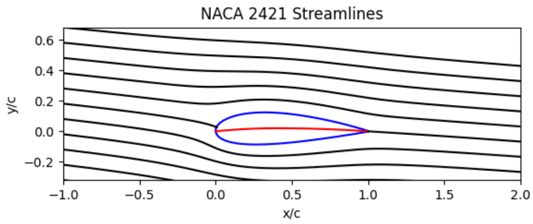
(b) NACA 2412 at 0 degrees



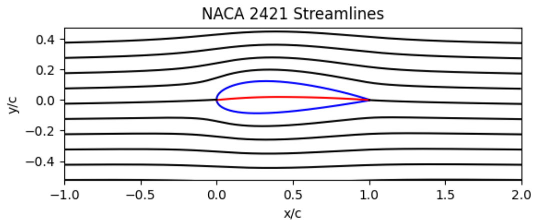
(c) NACA 2412 at 5 degrees

A18: Streamlines for NACA 2421 Airfoil

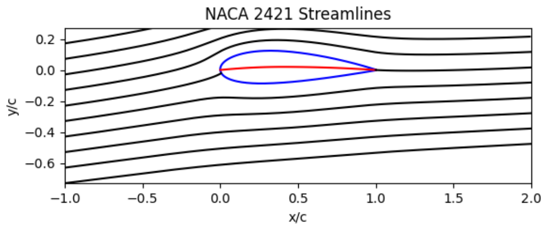
Use the code developed for problem A16 to plot the streamlines around the NACA 2421 airfoil at angles of attack of -5, 0, and 5 degrees.



(a) NACA 2421 at -5 degrees



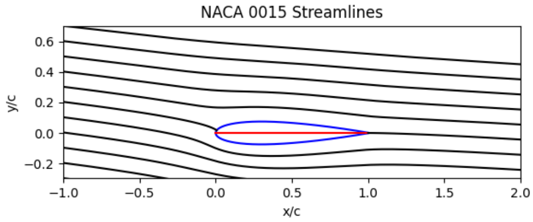
(b) NACA 2421 at 0 degrees



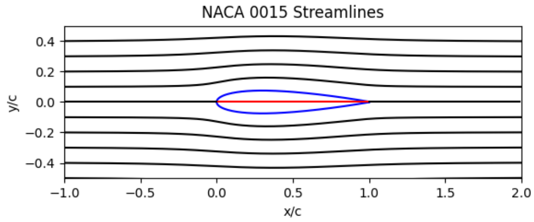
(c) NACA 2421 at 5 degrees

A19: Streamlines for NACA 0015 Airfoil

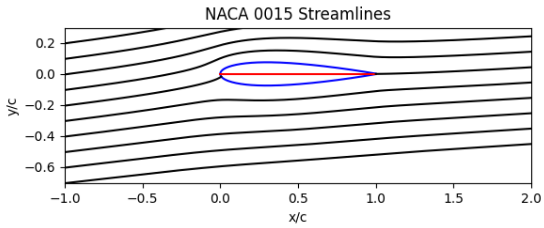
Use the code developed for problem A16 to plot the streamlines around the NACA 0015 airfoil at angles of attack of -5, 0, and 5 degrees.



(a) NACA 0015 at -5 degrees



(b) NACA 0015 at 0 degrees



(c) NACA 0015 at 5 degrees

A20: Comparison of Traditional Thickness and Closed Trailing Edge Thickness Distributions

Below is the comparison of the lift coefficient, leading-edge moment coefficient, and quarter-chord moment coefficient at zero degrees angle of attack for a NACA 2412 airfoil using both the traditional thickness distribution and the closed trailing edge thickness distribution:

Parameter	Traditional Thickness	Closed Trailing Edge Thickness	Percent Change
C_L	0.2610170742	0.2593304444	0.646 %
$C_{m,\mathrm{LE}}$	-0.1211471204	-0.1202041555	0.778 %
$C_{m(c/4)}$	-0.0558928519	-0.0553715444	0.933 %

Table 2: Comparison of C_L, $C_{m,\mathrm{LE}}$, and $C_{m(c/4)}$ for traditional and closed trailing edge thickness distributions, including percent change.

As seen from the table, the percent change in each parameter is relatively small, with the largest difference in the quarter chord moment coefficient.

A23: Pressure Coefficient on a NACA 4412 Airfoil

Use your code to plot the pressure coefficient on a NACA 4412 airfoil at zero degrees angle of attack with a closed trailing edge using 200 nodes and 199 panels. What is the lift coefficient?

CL = *0.5176742523081577*

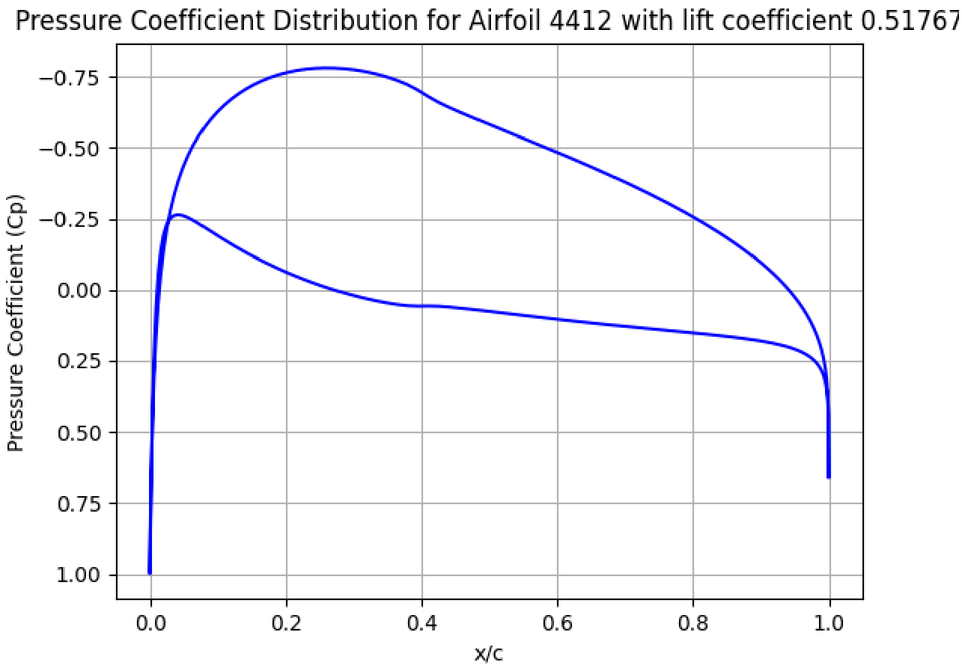


Figure 8: Pressure Coefficient vs Lift Coefficient - Section A23

A24: Pressure Coefficient on a Uniform-Load Airfoil

Use your code to plot the pressure coefficient on a uniform-load airfoil at zero degrees angle of attack with the same thickness distribution as was used in A23. What is the lift coefficient?

CL = 0.5418062510685514

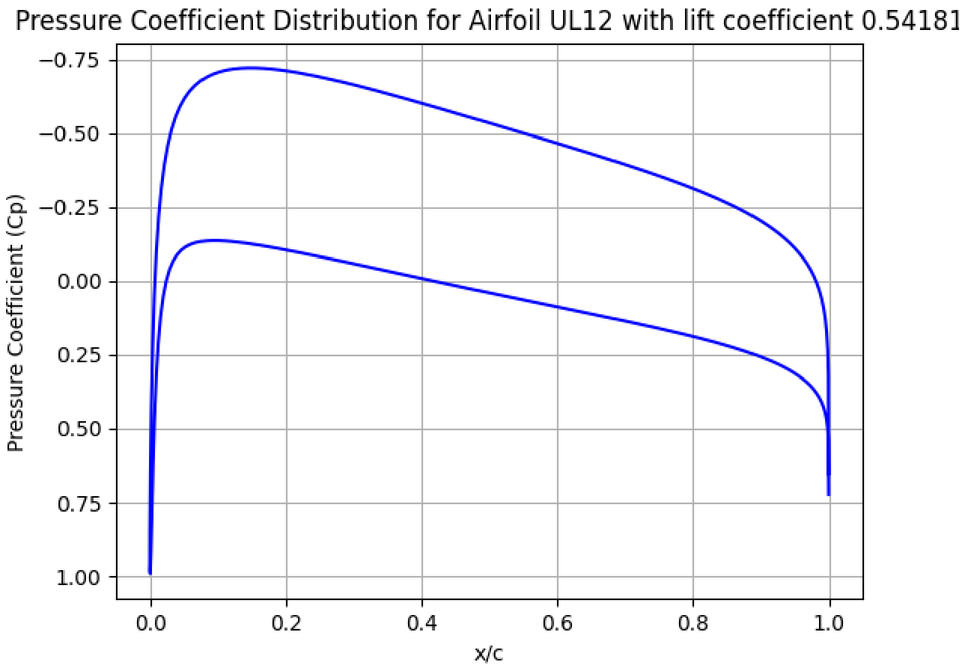


Figure 9: Pressure Coefficient vs Lift Coefficient - Section A24

A25: Pressure Coefficient with 1% Thickness

Use your code to plot the pressure coefficient on a uniform-load airfoil at zero degrees angle of attack with the same thickness distribution as was used in A23, but with only 1% thickness. What is the lift coefficient?

CL = 0.49406149921867325

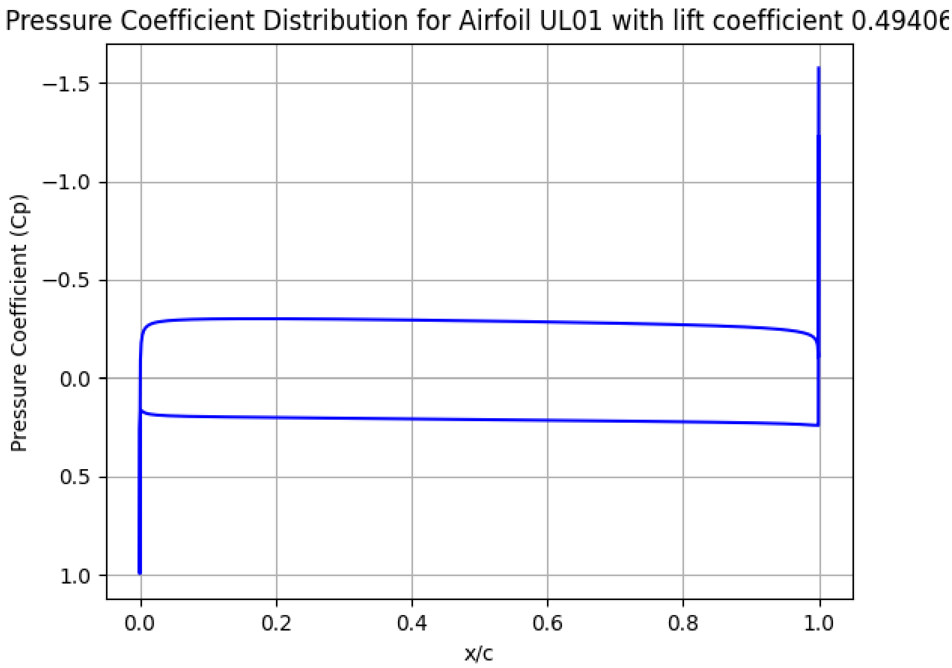


Figure 10: Pressure Coefficient vs Lift Coefficient - Section A25