

Airfoil Analysis

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Methods

An airfoil redirects the air flowing past it to generate lift force. When it does this, its motion in the direction of the airstream is resisted by drag force. Lift force and drag force are controlled by the lift and drag equations, shown in Equation 1:

$$\begin{aligned} L &= \frac{1}{2} \rho u^2 c_L \\ D &= \frac{1}{2} \rho u^2 c_D \end{aligned} \tag{1}$$

It can be seen that lift and drag force are both proportional to the square of velocity, but that they are subjected to different coefficients, the [Lift Coefficient](#) and the [Drag Coefficient](#). These coefficients change depending on the [Reynolds Number](#) of the wing and its [Angle of Attack](#), also denoted by the greek letter α . In addition to affecting the lift and drag experienced by an airfoil, the angle of attack can also affect its [Coefficient of Moment](#).

This project sought to identify how adjusting an airfoil's [Chord](#), or c , and [Camber](#) can affect its performance, making it better suited for different environments. For this project, I downloaded the coordinates of a variety of different airfoils from the [UIUC Applied Aeronautics Group](#) and then imported them to a julia program to calculate their [Lift Coefficient](#), [Drag Coefficient](#), [Pressure Drag Coefficient](#), and [Moment Coefficient](#). As can be seen from the included figures, these coefficients each changed with different angles of attack.

I used the [Xfoil.jl](#) Julia wrapper to calculate the coefficients of lift, drag, and . of the Xfoil potential flow software. Xfoil is a potential-flow code that can calculate an airfoil's lift, drag, pressure drag, moment coefficients. A [Potential-flow Code](#) divides the wing into many small plates and calculates the lift, drag, pressure drag, and moment coefficients for each of a finite number of points along the length of the wing. This program is repeated with different starting estimates for each coefficient until it either converges or is run a set number of times. The user can adjust the limit of how many times the code can run before it gives up.

The [julia file](#) created for this project and its accompanying [header file](#) could control separately for different input options. I included one function that used a default angle step size of 1 degree, with 100 iterations. It found the lowest and highest angles for which a solution converged, then plotted all four solutions overlaid on each other between these angles. A second function allowed the user to adjust the number of iterations, and a third function allowed the user to manually set the maximum and minimum angles of attack.

For this code, it also became necessary to create similar airfoils, only adjusted slightly for the size and position of their [Camber](#), and the maximum thickness as a percentage of the [Chord](#). To do this, I created a function in the previously mentioned [header file](#) to calculate the shape of an airfoil based on its [NACA Airfoil](#) (National Advisory Committee for Aeronautics) number. Although there are also other ways to describe an airfoil's profile, this seemed like a clear and efficient method.

Each of the digits in a [NACA Airfoil](#) has its own purpose in defining an airfoil. The first digit is the ratio of the maximum [Camber](#) to the [Chord](#), multiplied by ten. The second digit is the distance of the location of maximum camber from the leading edge of the wing, in tenths of the entire length of the wing. The third and fourth digit define the maximum wing thickness as a percentage of the Chord. The profiles of both the top and bottom of the wing can be described by Equations 2 and 3. In these Equations, x represents the distance from the front of the wing, y_t represents the wing thickness at x , y_c represents the location of the wing's center at x , t represents the maximum thickness as a decimal percentage of the [chord](#), m denotes the wing's maximum camber as a percent of the wing's cord length, and p is the distance of the maximum camber from the front of the wing, in tenths.

$$y_c = \frac{m}{p^2}(2px - x^2) \quad 0 \leq x \leq p$$

$$y_c = \frac{m}{(1-p)^2}((1-2p) + 2px - x^2) \quad p \leq x \leq 1 \quad (2)$$

$$y_t = 5t[0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1015x^4] \quad (3)$$

The code to create an airfoil from its [4-digit NACA Number](#) was first checked against an airfoil file downloaded from the [UIUC Applied Aeronautics Group](#) website. Once these were found to be the same, this code was used to create similar airfoils that only differed slightly in either their maximum camber, maximum camber location, and maximum wing thickness. These results were then compared to measure the effect of each variable.

Results and Discussion

This research found that adjusting the Reynolds number, angle of attack, thickness, and camber amount and position all affect the forces experienced by an airfoil. These forces are described by Equation 1 on the first page. Figure 1 and Figure 4 show that airfoils with a larger [Camber](#) experience more lift force. The airfoils shown in Figure 2 and Figure 5 experience the same increase in [Drag Coefficient](#) as they increase their Camber-to-Cord ratio. Both coefficients increased with increasing angle of attack. On the range from -8° to 8° shown in the graphs, the Drag Coefficient increased as angle of attack became either more positive or more negative.

Table 1 contains a list of all the airfoil NACA numbers and Reynolds numbers used in this lab. Clicking on the [hyperlink number](#) below each NACA number will jump to the plot of its lift, drag, or moment coefficients compared to angle of attack.

One difficulty that was encountered during the analysis of these airfoils was making smooth graphs for them over their entire angle range. This was difficult because the airfoils compared were sometimes significantly different. For example, Airfoils 1 and 2 had very close [NACA Numbers](#), with the only difference between them being their first digit. This 1-digit difference meant, though, that Airfoil 2 had twice as much [Camber](#). This meant that although their behaviors could be plotted on the same plane in Figures 1, 2, and 3, they were very different. This made it difficult to plot converging solutions for the different coefficients of these wing types on the same plane.

Table 1: Airfoils Modeled by Program

A link to the graph for each airfoil is in its bottom row.

Profile	1430	2430	3430	1210	2210	3210	2230	2230	2230
<i>Re</i>	10,000	10,000	10,000	10,000	10,000	10,000	10,000	20,000	30,000
Link	1	2	3	4	5	6	7	8	9
Profile	1120	1120	1120	2230	2235	2240	2120	2220	2320
<i>Re</i>	10,000	20,000	30,000	10,000	10,000	10,000	10,000	10,000	10,000
Link	10	11	12	13	14	15	16	17	18

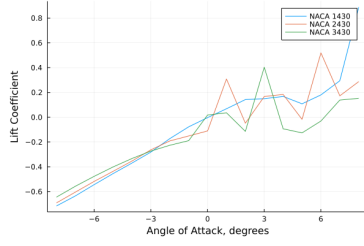


Figure 1: Lift Coefficient Compared to Varying Camber Length

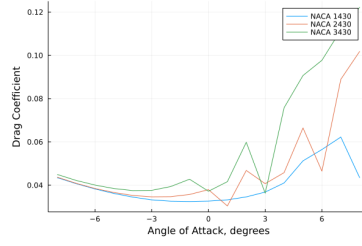


Figure 2: Drag Coefficient Compared to Varying Camber Length

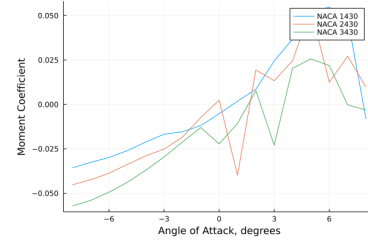


Figure 3: Moment Coefficient Compared to Varying Camber Length

The plots in Figure 1 and Figure 4 appear to have a similar upward slope, which is expected. As the wing tilts upwards with an increasing angle of attack its Lift Coefficient increases from negative to positive for both airfoils. A notable difference, though, is that for the airfoils with their maximum camber further forward shown in Figure 5 had a much greater drag coefficient at lower angles than those in Figure 2.

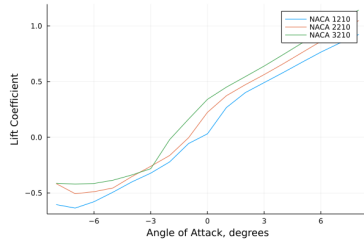


Figure 4: Lift Coefficient Compared to Varying Camber Length

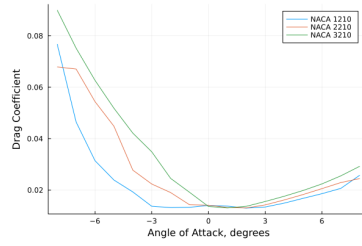


Figure 5: Drag Coefficient Compared to Varying Camber Length

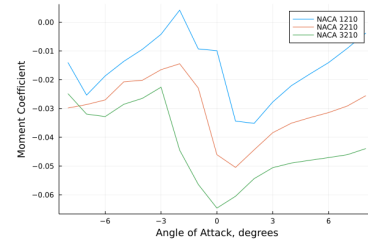


Figure 6: Moment Coefficient Compared to Varying Camber Length

Adjusting the Reynolds Number of an airfoil also changed its behavior significantly. The unitless Reynolds number of an airfoil, described by Equation 4, can change for an aircraft even as its velocity or the air density changes.

$$Re = \frac{uL}{\nu} = \frac{\rho uL}{\mu} \quad (4)$$

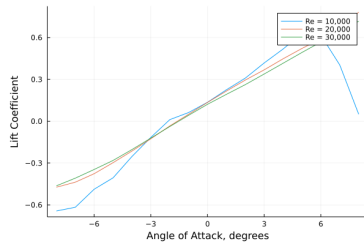


Figure 7: NACA 2230 Lift Coefficients Varied by Reynolds Number

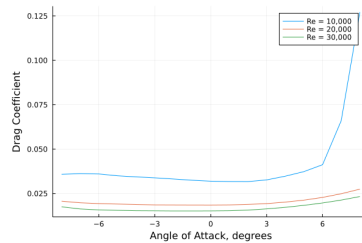


Figure 8: NACA 2230 Drag Coefficients Varied by Reynolds Number

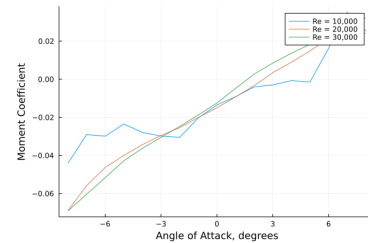


Figure 9: NACA 2230 Moment Coefficients Varied by Reynolds Number

Both the NACA 2230 Airfoil shown in Figure 7 and the NACA 1120 Airfoil in Figure 10 appear to increase in **Lift Coefficient** more rapidly at lower Reynolds Numbers. Their **Drag Coefficients** are seen to be higher at low Reynolds Numbers in both Figure 8 and Figure 11. At low Reynolds numbers their **Moment Coefficients** are both higher, and then either increase more slowly or decrease more quickly.

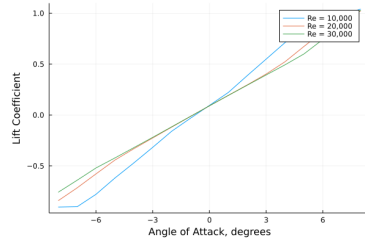


Figure 10: NACA 1120 Lift Coefficients Varied by Reynolds Number

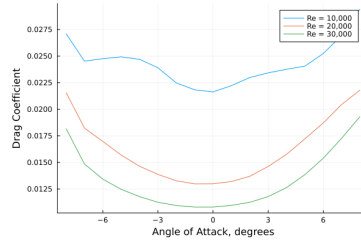


Figure 11: NACA 1120 Drag Coefficients Varied by Reynolds Number

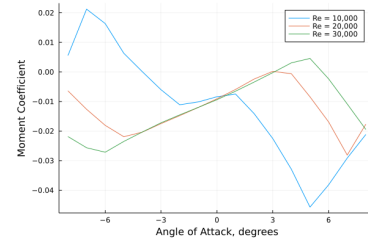


Figure 12: NACA 1120 Moment Coefficients Varied by Reynolds Number

Figure 13 shows that varying the **Wing Thickness** only slightly can have huge impacts on the Lift, Drag, and Moment of an airfoil. In the case of Lift and Moment, a few percent difference in thickness can even change its sign or slope. A wing's thickness varies the position of its surface at all points along its profile, which add up to flow to be redirected in significantly different ways.

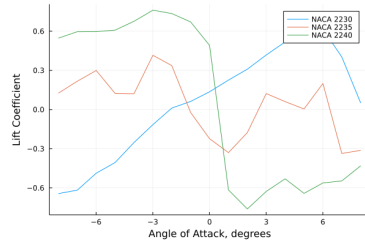


Figure 13: Lift Coefficients Varied by Maximum Camber Location

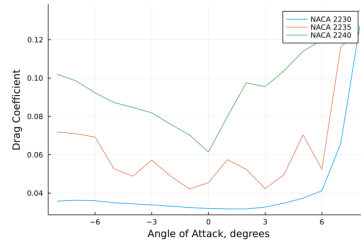


Figure 14: Drag Coefficients Varied by Maximum Camber Location

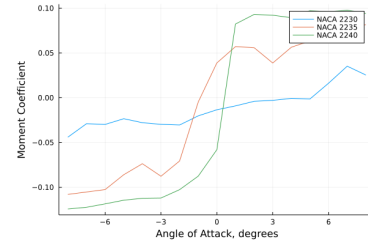


Figure 15: Moment Coefficients Varied by Maximum Camber Location

Although **Wing Thickness** has been shown to significantly impact an airfoil's Lift, Drag, and Moment Coefficients, Figure 16 shows that the position of the **Maximum Camber** does not seem to significantly impact the Lift Coefficient. Figure 17 shows that drag coefficient is hardly impacted either. Moment coefficient is perhaps changed the most, shown in Figure 18, but this is still much less than the change caused by wing thickness shown in Figure 15.

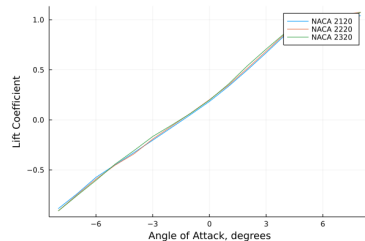


Figure 16: Lift Coefficients Varied by Maximum Camber Location

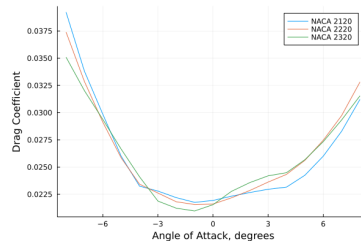


Figure 17: Drag Coefficients Varied by Maximum Camber Location

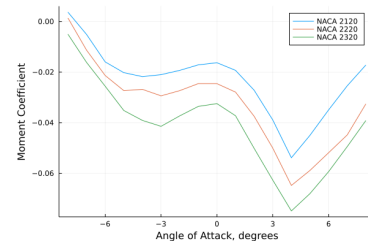


Figure 18: Moment Coefficients Varied by Maximum Camber Location

I found in this airfoil analysis that an airfoil experiences different [Lift](#), [Drag](#), and [Moment](#) forces at different [Angles of Attack](#). I found that in general, Lift force increases with increasing angle of attack, and drag force behaves as a parabola with a minimum near zero. Zooming out further beyond the 16-degree domain of these graphs would show that these coefficients actually behave as sinusoidal waves as the airfoil's angle of attack moves around in a circle.

A surprising finding was that the Lift and Drag coefficients experienced by an airfoil are not significantly impacted by its maximum camber location, as seen in Figure 18. It must not actually impact where airflow is actually redirected. Thicker wings appear to experience greater Drag and lower Lift, as seen in Figures 13. and 14. A higher Reynolds Number appears to correspond to a faster increase in the [Lift Coefficient](#), but also a much higher [Drag Coefficient](#).

For this project, I also completed the example in the [Xfoil.jl Documentation](#) provided with the lab, using [this code](#).

1 Glossary

- Angle of Attack, α - The angle of attack, α is the angle between the motion of oncoming fluid and the chord line of the wing. a positive α corresponds to a wing tilted upwards.
- Camber - The Camber of an airfoil is represented by the Camber Line, which runs halfway between its top and bottom surfaces. This line represents the curvature of an airfoil. An airfoil with positive camber is slightly convex on top and slightly concave on its bottom.
- Chord, c - An airfoil's Chord is the imaginary line running straight from the tip of an airfoil to its tail. The chord line is used to find an airfoil's [Angle of Attack](#).
- Drag Coefficient (2D), c_D - The drag coefficient determines how much drag force opposing motion will be experienced. It comes from a combination of [Pressure Drag](#), also called Form Drag, and [Viscous Drag](#) or Skin Friction. The drag force equation describes it in this way:

$$D = \frac{1}{2}\rho u^2 c_D \quad (5)$$

- Freestream Velocity, V_∞ - The velocity of an oncoming air flow directly upstream from an airfoil, before it interacts with it.
- Lift Coefficient (2D), c_L - The lift coefficient is used in the equation below to define how much lift force acts perpendicular to the direction of the oncoming fluid flow.

$$L = \frac{1}{2}\rho u^2 c_L \quad (6)$$

- Lift Curve Slope - The lift curve plots the [Lift Coefficient](#) against the [Angle of Attack](#). This shows the effect that changing the angle of attack will have on the plane's total lift force.
- Mach Number, M - The Mach number is the velocity of an object in proportion to the speed of sound in its medium. When an airfoil is traveling near the speed of sound, its top portions can have fluid velocity above Mach 1. When an airfoil is traveling above Mach 1, the fluid on both sides of it also have velocities above mach 1 while there are points directly before and after it that are below Mach 1.

$$M = \frac{u}{c} \quad (7)$$

- Pressure Drag - Pressure Drag, also called Form Drag, comes from the the formation of a vacuum behind an object. The object experiences higher pressure ahead of it than behind it, so the pressure difference pushes it backwards.
- Pitching Moment Coefficient (2D), c_M - The moment coefficient is used to calculate the pitching moment a wing will experience from its dynamic pressure q , area S , and chord length c .

$$M = qSc c_M \quad (8)$$

- Polar - The Airfoil Polar is a plot showing the [Lift](#) and [Drag](#) coefficients corresponding with each [Angle of Attack](#) for an airfoil. Examining the ratios of lift to drag is instrumental in choosing the optimal Angle of Attack.
- Potential-Flow Code - A potential-flow code calculates the [lift](#), [drag](#), and [moment](#) forces an airfoil experiences at many different points along it. Potential-flow theory assumes constant, incompressible, inviscid fluid flow. It calculates the coefficient for each of these small points and then combines them to find the coefficients along the entire wing.

- Reynolds Number, Re - The Reynolds Number is a unit-less number for fluid flow described by the equation below. It can be used to predict patterns in the fluid's flow, using its flow speed u , characteristic length L , and kinetic viscosity ν , or else by its density ρ , flow speed u , characteristic length L , and fluid density μ .

$$\begin{aligned} Re &= \frac{uL}{\nu} \\ &= \frac{\rho uL}{\mu} \end{aligned} \tag{9}$$

- Stall - Stall occurs when an airfoil's [Angle of Attack](#) is too great in magnitude. When the angle of attack is too great, flow separation occurs, reducing rather than augmenting the airfoil's [Lift Coefficient](#) as the Angle of Attack increases.
- Thickness - An airfoil's thickness can be measured in two different ways, either along its [Chord Line](#) or along its [Camber Line](#). Thickness measured perpendicular to the Camber Line is also called the American Convention, and thickness measured perpendicular to the Chord Line is also called the British Convention.
- Viscous Drag - Viscous Drag, also called Skin Friction, is drag caused by friction with the fluid particles flowing past an airfoil. Along with [Pressure Drag](#), it contributes to an airfoil's total [Drag Coefficient](#) airfoil.
- 4-Digit NACA Airfoil - A wing shape developed by the National Advisory Committee for Aeronautics (NACA). The first digit is the maximum [Camber](#) in tenths of the [Chord](#). The second digit is the distance in tenths of the maximum Camber from the leading edge, out of ten. the final two digits are the maximum wing thickness as a percentage of the Chord. Descriptions of other NACA numbers can also be found on this [Wikipedia Article](#).