

A Simulation-Based Experimental Investigation into Morphing Wings

Research Question: To what extent can the adoption of morphing camber wings help reduce parasitic drag and improve the stall characteristics of aircraft compared to conventional wings with flaps?

Table of Contents

1: Introduction.....	3
1.1 Abstract.....	3
1.2 Literature Review.....	4
2: Background Information.....	5
2.1 Boundary Layer.....	5
2.2 Flow separation and Pressure Drag.....	6
3: Experiment.....	8
3.1 Methodology.....	8
3.1.1 RANS Equations.....	8
3.1.2 Eddy Viscosity Turbulence Models.....	9
3.1.3 Computational Fluid Dynamics.....	10
3.2 Variables.....	10
3.2.1 Dependent Variables.....	11
3.2.2 Independent Variables.....	11
3.2.3 Controlled Variables.....	13
3.3 Setup.....	13
3.3.1 Airfoil geometries.....	14
3.3.2 CFD parameters.....	14
3.3.3 Conducting the experiment.....	15
4: Data Analysis.....	16
5: Conclusion.....	26
5.1 Limitations of Approach.....	26
5.2 Efforts to Reduce Error.....	26
5.3 Summary of results.....	27
References.....	29
6: Appendix.....	31

1: Introduction

1.1 Abstract

Morphing camber airfoils are a novel wing technology that forgo the need for traditional, mechanically complex, and aerodynamically inefficient wing control surfaces such as flaps by morphing the entire wing body, thus altering its aerodynamic properties like flaps do, but in a more aerodynamically efficient manner. More specifically, morphing camber airfoil technology morphs a wing's camber, which refers to how "curved" a wing's airfoil is. In this research study, the claim that morphing camber wings are more aerodynamically efficient while also having better stall characteristics than conventional wings with flaps was tested. In order to do this, the study conducted CFD based simulations comparing the aerodynamic performance of conventional and morphing camber airfoils using CFD software Ansys Fluent. The data from the simulations revealed that on average, morphing camber airfoils have a 19.7% higher Lift-to-Drag ratio across various morphing rates and α . Additionally, it was found that morphing cambered airfoils also stalled gradually with increasing α , instead of stalling abruptly like conventional airfoils, which is a more desirable stall characteristic. Further three dimensional and higher fidelity (URANS/DES/LES) studies are needed to confirm these results for full wings.

1.2 Literature Review

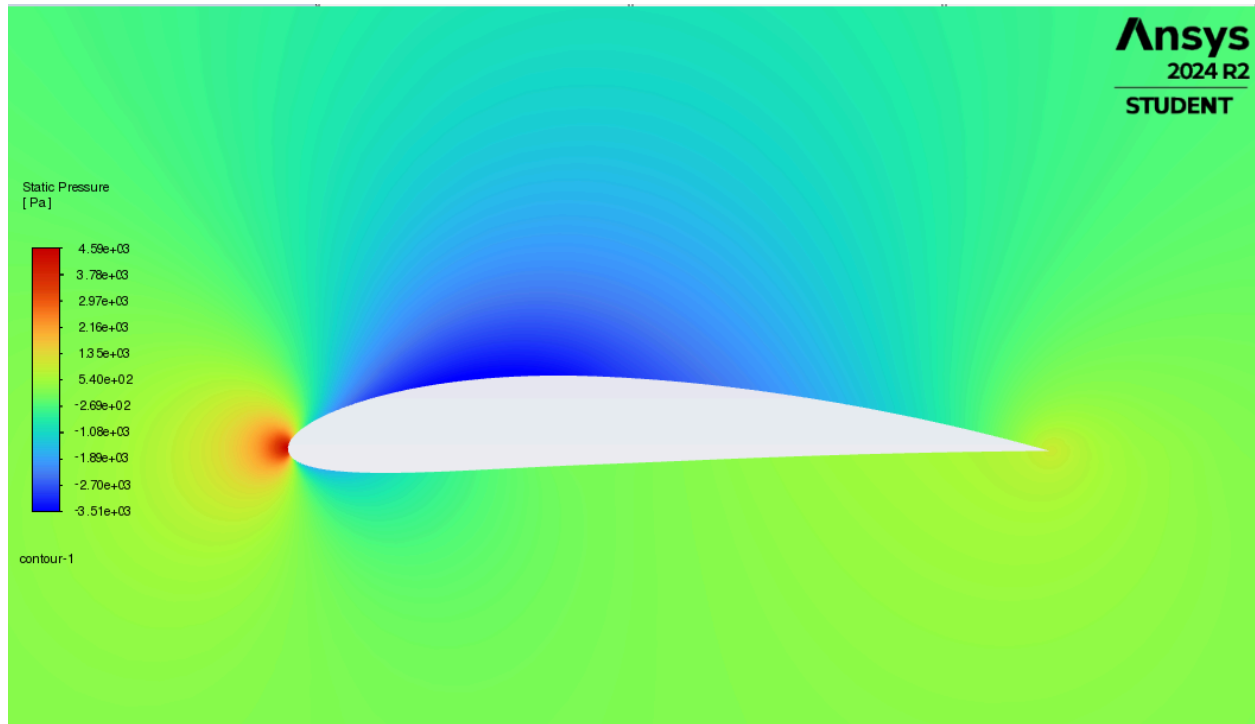
Many studies have been done on the structural and engineering challenges surrounding morphing airfoils [1]. However, little literature exists to compare the actual aerodynamic benefits of this new wing technology. Hence, this study will attempt to find out to what extent morphing wings are actually more aerodynamically efficient than conventional wings, as well as if there are any improvements in the stall characteristics. Although Majid et al have compared morphing airfoils to conventional airfoils using CFD software before, they compared morphing airfoils against conventional airfoils with plain flaps, not the more modern, more efficient, and more widely used slotted flaps [2]. They also used a less accurate turbulence model, the Spalart–Allmaras model, instead of the $k-\omega$ SST model [7], which is even more important considering the turbulent flow due to the slotted flaps and flow separation at high α . Additionally, the Reynolds number that they used does not reflect the Reynolds number found in most real world applications. A Reynolds number of 76,000 translates approximately to only 1.5 m/s airspeed. Therefore a more appropriate and more common Reynolds number of 6 million was chosen for this study.

2: Background Information

2.1 Boundary Layer

Suppose air is flowing around an airfoil at a velocity defined as the free stream velocity. At the very surface of the airfoil, the first layer of air molecules adhere to the surface molecules due to the no-slip condition. Due to intermolecular interactions (viscous effects) and momentum transfer, the velocity of the next layer of air molecules above is also slowed down, but faster than the initial layer. As distance from the airfoil surface increases, the velocity of the air molecules increases until a certain distance away, the air molecules are travelling at the freestream velocity again. This transition region between the surface of the airfoil and the free stream air is known as the boundary layer [3]. The boundary layer can be laminar or turbulent. In a turbulent boundary layer, there is vigorous mixing of airflow between the different layers, which increases its skin friction coefficient. However, also due to the mixing, the average velocity of the air within a turbulent boundary layer is higher.

2.2 Flow separation and Pressure Drag



^[10]Figure 1

Figure 1 shows a pressure contour diagram of airflow around a NACA 4412 airfoil at $\alpha = 0^\circ$. It can be observed that as the airflow travels over the surface of an airfoil, it passes through different pressure regions. As the air travels from just aft of the leading edge towards the trailing edge of the airfoil, it encounters an adverse pressure gradient. In moving across this adverse pressure gradient, the air loses some of its velocity. This can become a problem if the air in the boundary layer loses all of its velocity and begins to travel in the opposite direction of the freestream velocity. When this occurs, the boundary layer detaches from the airfoil and this is also known as flow separation [3]. A consequence of flow separation is that it produces a region known as a wake behind the point at which the flow separation occurs. The wake is a region of

recirculating air with many eddies. The energy of the eddies is eventually lost to viscous forces in the air. As a result of the loss in energy, the pressure of the air in this region also decreases, this can be observed in figure 1 as well. Since the leading edge of the airfoil experiences higher pressure due to the stagnation point, while the trailing edge experiences lower pressure due to the wake, a pressure difference is created between the front and back of the airfoil. The pressure difference creates a force that pushes the airfoil back, which is also known as pressure drag [4].

3: Experiment

3.1 Methodology

3.1.1 RANS Equations

$$\nabla \cdot \mathbf{u} = 0 \quad (1)$$

$$\rho \frac{d\mathbf{u}}{dt} = -\nabla p + \mu \nabla^2 \mathbf{u} + \mathbf{F} \quad (2)$$

The RANS Equations stands for Reynolds Averaged Navier Stokes Equations. As the name implies, they are a time-averaged formulation of the Navier Stokes (NS) equations. The incompressible NS equations (equations 1 and 2) are a set of Partial Differential Equations that can predict the behaviour of fluids, given a set of parameters [5]. Although the NS equations are able to predict the behaviour of a fluid accurately given the correct parameters, solutions tend to quickly become chaotic, making it extremely difficult or impossible to calculate a direct analytical solution using supercomputers for even the simplest problems. However, if some assumptions and simplifications are made, it is possible to achieve acceptable accuracy for a

fraction of the computational cost using numerical methods. This is what the RANS equations (equation 3) do.

$$\frac{\partial(\rho U_i)}{\partial t} + \frac{\partial(\rho U_i U_j)}{\partial x_j} = -\frac{\partial P}{\partial x_i} + \frac{\partial}{\partial x_j} \left[\mu \left(\frac{\partial U_i}{\partial x_j} + \frac{\partial U_j}{\partial x_i} \right) - \rho \overline{u'_i u'_j} \right] \quad (3)$$

The main difference between the NS and RANS equations is that the RANS equations are time-averaged. Through a technique called Reynolds decomposition, an instantaneous quantity is decomposed into a time-averaged and fluctuating component. The fluctuating component is accounted for by the Reynolds stress term [6].

3.1.2 Eddy Viscosity Turbulence Models

A separate turbulence model is required to solve for the Reynolds stress term. For this study, the k- ω SST eddy viscosity turbulence model (equations 4 and 5) was used [7]. Eddy viscosity turbulence models, as their names suggest, utilize the fact that increasing the turbulence in a fluid has a similar effect as increasing the viscosity of a fluid. This is due to a phenomenon known as the Boussinesq hypothesis, which states that eddies in a fluid cause momentum transfer between the molecules in a fluid, similar to how stronger intermolecular forces present in more viscous fluids also cause momentum transfer between the molecules in a fluid [8]. By utilizing this generalization, eddy viscosity models are able to bypass calculating for every individual eddy caused by the turbulence, saving a great deal of computational cost.

$$\frac{\partial(\rho k)}{\partial t} + \frac{\partial(\rho u_j k)}{\partial x_j} = P - \beta^* \rho k \omega + \frac{\partial}{\partial x_j} \left[(\mu + \sigma_k \mu_t) \frac{\partial k}{\partial x_j} \right] \quad (4)$$

$$\frac{\partial(\rho \omega)}{\partial t} + \frac{\partial(\rho u_j \omega)}{\partial x_j} = \gamma \frac{P}{\nu_t} - \beta \rho \omega^2 + \frac{\partial}{\partial x_j} \left[(\mu + \sigma_\omega \mu_t) \frac{\partial \omega}{\partial x_j} \right] + 2(1 - F_1) \rho \sigma_{\omega 2} \frac{1}{\omega} \frac{\partial k}{\partial x_j} \frac{\partial \omega}{\partial x_j} \quad (5)$$

3.1.3 Computational Fluid Dynamics

The aforementioned RANS equations and the accompanying Eddy Viscosity Turbulence Models are solved by a class of computer programs known as Computational Fluid Dynamics (CFD) software. These programs take in geometries and parameters that the user wants to simulate for, and use the numerical methods mentioned above to solve for an approximate solution [9]. For this study, the CFD software Ansys Fluent was used [10]. All simulations were performed assuming steady, incompressible flow with fully turbulent boundary layers.

3.2 Variables

3.2.1 Dependent Variables

Lift Coefficient (C_L)

The C_L is a dimensionless quantity that relates the air velocity (V), entire wing area (A), and density of the air (ρ) to the lift (L) produced by a wing. The C_L is proportional to lift and is often used in place of lift as it allows for a more generalized and scalable comparison to other airfoils and flow conditions.

$$C_L = \frac{L}{\frac{1}{2}\rho V^2 A} \quad (6)$$

Drag Coefficient (C_d)

The C_d is a dimensionless quantity that relates the velocity, wing area, and density of the air to the drag produced by a wing. The C_d is used in place of drag for the same reason as the C_L .

$$C_D = \frac{D}{\frac{1}{2}\rho V^2 A} \quad (7)$$

3.2.2 Independent Variables

Camber rate

Camber is a measure of how curved an airfoil is. Camber rate is a measure of how cambered an airfoil is. In the NACA 4-digit naming system (NACA XXXX), camber is determined by the first number. For example, in a NACA 2XXX airfoil, 2% of the chord length is the max camber. This experiment will vary the camber rate from 0% (NACA 0012), in increments of 2%, till 8% (NACA 8412) to simulate varying morphing stages of a camber morphing airfoil. Increasing the camber of an airfoil generally increases the lift it produces at any given α .

Flap deflection angle (°)

The flap deflection angle is a measure of how many degrees the slotted flap on the conventional airfoil is deflected from its original position. This experiment will test 5 different flap deflection angles for the conventional airfoil as matching cases for the 5 camber rates for the morphing airfoils. Increasing the flap deflection angle generally increases the lift the airfoil produces at any given α .

Angle of Attack / α (°)

α is a measure of how many degrees the chord line of an airfoil is angled away from the oncoming freestream air. Increasing the α generally increases the lift produced by an airfoil, to a certain extent, after which a stall is likely to occur, causing a sharp decrease in lift.

3.2.3 Controlled Variables

Chord length (m)

The chord length of an airfoil is the distance between the leading edge and the trailing edge of an airfoil. The chord length was kept constant at 1 meter for all tested airfoils.

Reynolds number / Re

The Reynolds number is dimensionless quantity that is dependent on several quantities such as the density of the air, the freestream velocity, the chord length, and dynamic viscosity of the air. For this study, a Reynolds number of 6 million was chosen as it is a more accurate reflection of the conditions experienced by most aircraft. By inputting the values of constants such as $\rho = 1.225 \text{ kg/m}^3$, $L = 1 \text{ m}$, $\mu = 1.802 \times 10^{-5}$, we are able to solve for the freestream velocity, which is equal to 88.3 m/s.

$$Re = \frac{\rho V L}{\mu} \quad (8)$$

3.3 Setup

3.3.1 Airfoil geometries

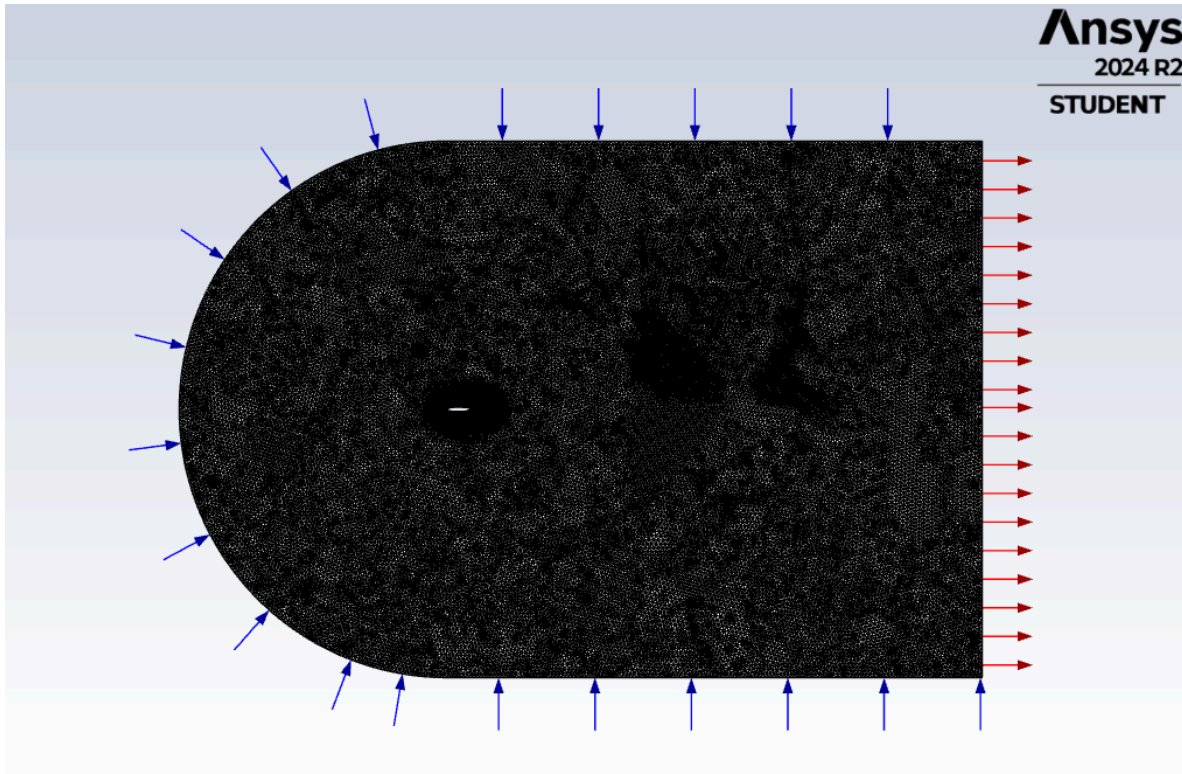
Morphing airfoil geometries were created by importing csv coordinate files of airfoils from airfoilplotter directly into Ansys Designmodeler. The conventional airfoil geometry used was the popular NACA 23012 airfoil with a slotted flap [13]. The airfoil geometry was traced from a drawing, then imported into Computer Aided Design (CAD) software SolidWorks [14], where flap deflection angles were changed.

3.3.2 CFD parameters

A 2D simulation was started using Ansys Fluent. Steady, incompressible flow with fully turbulent boundary layers were assumed for all simulations. A C-shape fluid domain with dimensions: 7.5 m radius, 15 m height, 15 m length seen in Figure 2 was used for all simulations, as suggested by Thomas [11]. For the mesh, an edge sizing of 200 divisions was used around the surface of the airfoil. Triangular elements were set as the default element shape. Next, 25 inflation layers with a growth rate of 1.1 were used, similar to Majid et al in order to ensure that the wall Y^+ value was under 1 so that the viscous sublayer could be properly resolved [2]. An element size of 0.1m was chosen. Approximately 90,000 elements were present in the final mesh. For the setup of the fluent solver, a pressure based solver coupled with the $k-\omega$ SST turbulence model was used. The convergence criteria for residuals was set to 1×10^{-6} as suggested

by Thomas [11]. Each simulation was iterated until all C_d / C_L / residuals values exhibited stable convergence, with a minimum of 300 iterations performed per case.

Figure 2: Representative fluid domain and mesh used for all simulations, consisting of approximately 90,000 elements.



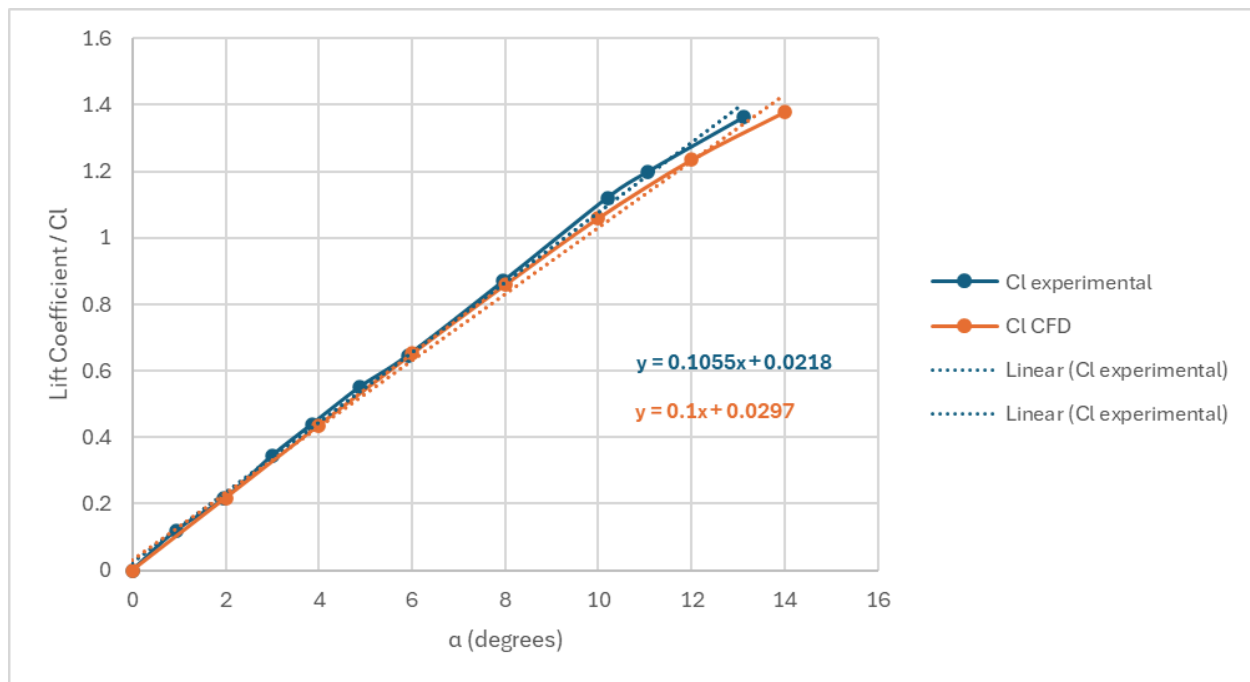
3.3.3 Conducting the experiment

Initially, 5 camber rates (0%, 2%, 4%, 6%, 8%) were simulated for, each at 8 different α (0° , 2° , 4° , 8° , 10° , 12° , 14°). Then, 5 matching cases for the initial C_L values of the morphing airfoils at an $\alpha = 0^\circ$ were found for the conventional airfoils with slotted flaps. The 5 matching case flap deflection angles were (-0.25° , 2.5° , 5.75° , 9.5° , 12.75°). Later, 3 additional α (16° , 18° , 20°)

were simulated for the 8% camber morphing airfoil (NACA 8412) and its matching case conventional airfoil with flap deflection of 12.75° in order to see the behaviour of the airfoils at even higher α .

4: Data Analysis

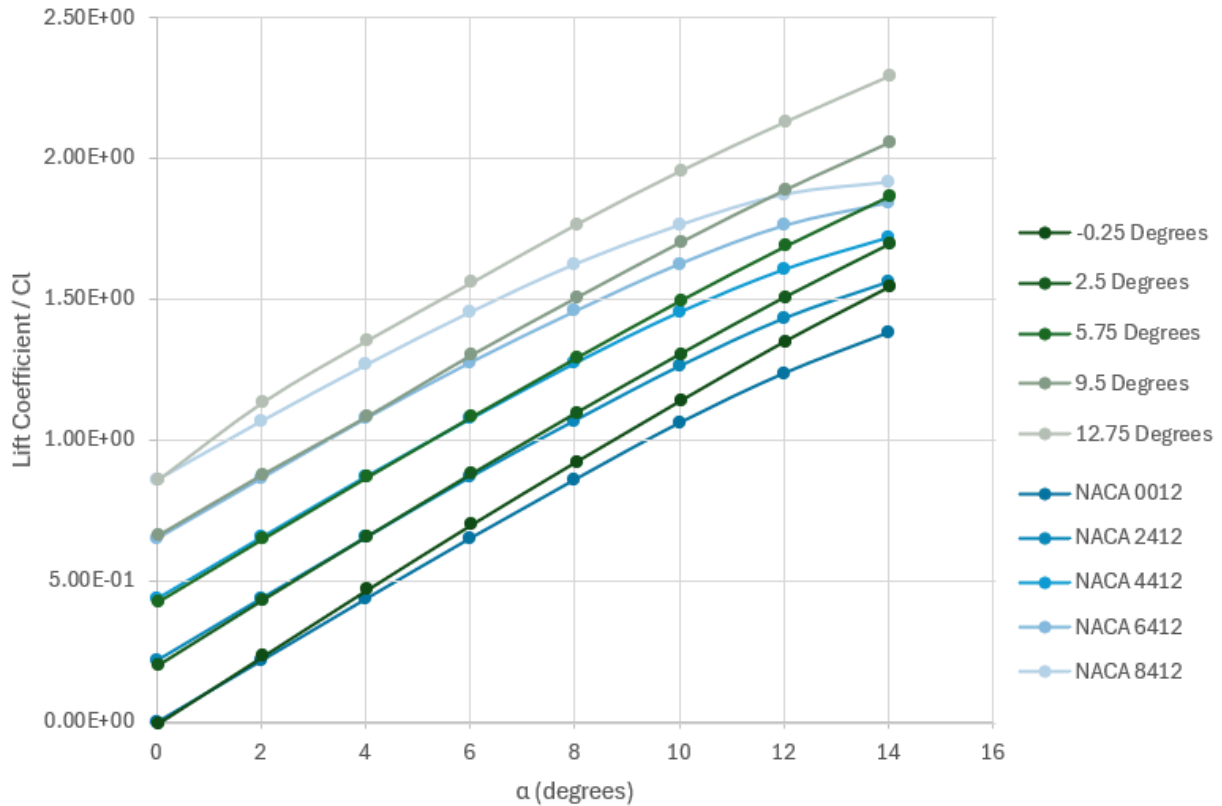
Figure 3: Experimental Validation Case Comparing Experimental Data to CFD Data (NACA 0012 at $Re = 6$ million.)



In order to test the accuracy of CFD software in simulating the airflow around airfoils, an experimental validation study was conducted. Experimental C_L data for a NACA 0012 airfoil at varying α and a Reynolds Number of 6 million was obtained from Abbott et al [12]. This set of data was graphed alongside C_L data obtained using Ansys Fluent CFD software. As can be seen in Figure 3, the trends produced by each dataset are very similar to each other. Comparing the

slopes, the error is approximately only 5.21%. This is an acceptable amount of error considering the many assumptions and simplifications made by the CFD software. Hence, the same parameters and setup was used for the rest of the simulations.

Figure 4: CFD Data for C_L vs α for Various Morphing Rates and α at $Re = 6$ million.



Matching starting cases were obtained for each morphing airfoil camber rate. For example, morphing the camber rate by 2% produced the same lift as a flap deflection of 2.5° at $\alpha = 0^\circ$. Morphing the camber rate by 4% produced the same lift as a flap deflection of 5.75° . These matching cases were found by varying the flap deflection angle in 0.25° increments until the closest match to a morphing airfoil's C_L was found. Perfect matching cases were not sought after as this would require even finer increments in the flap deflection angle, which would be overly

time consuming. As a result, there is a slight error in the matching cases as presented in Table 1. The percent error is relatively low for most cases, except for an outlier in the case of the morphing 0% camber airfoil due to the 0.25° increments not being precise enough.

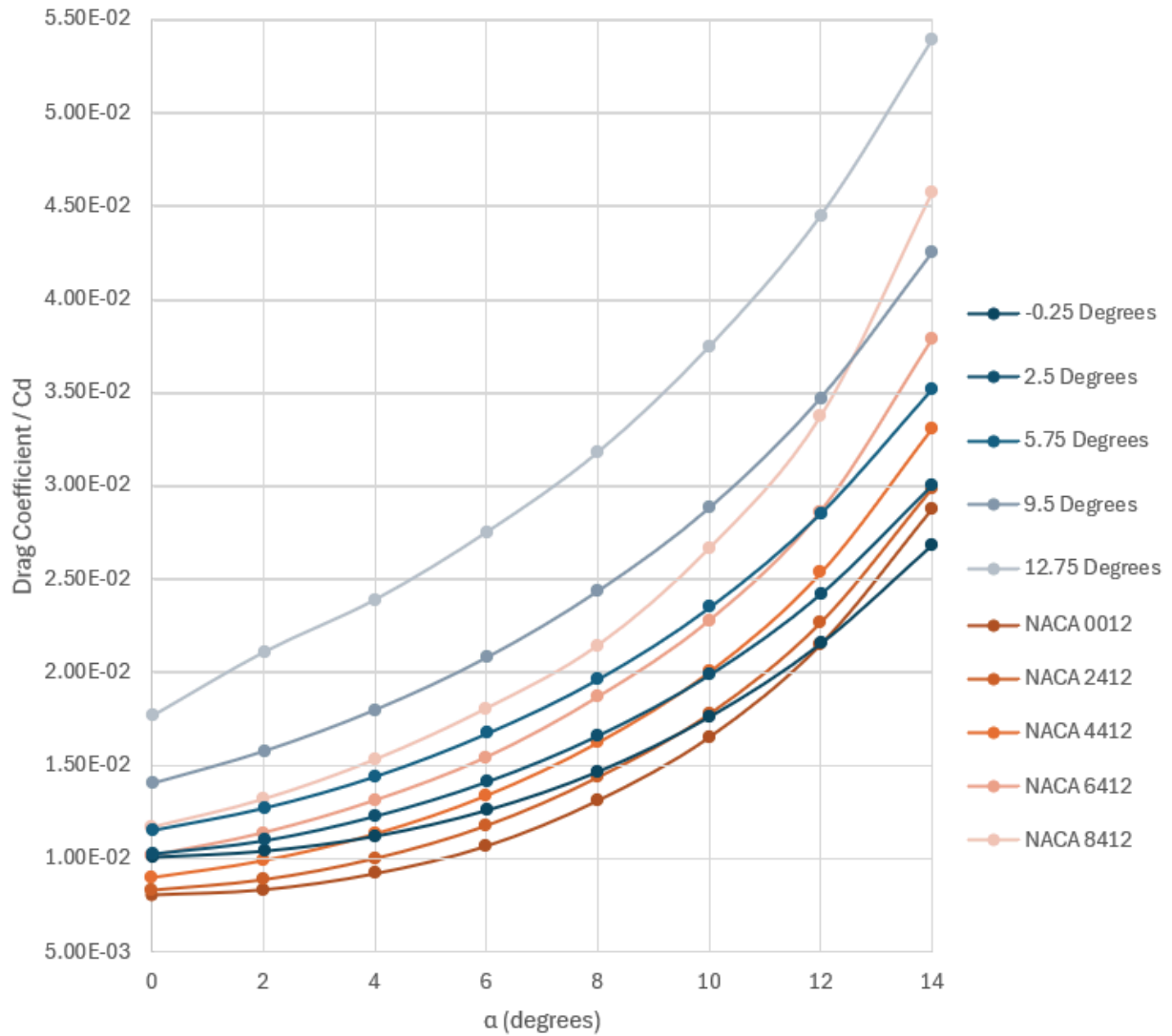
It can be observed that the conventional airfoils produce greater lift than the morphing airfoils at almost all camber rates and α . The difference grows larger with increasing α and the rate at which the difference grows also increases with greater camber rates. This can be attributed to greater flow separation in the morphing airfoils than conventional airfoils. Flow separation modifies the apparent shape of the airfoil to the oncoming air, effectively reducing the α despite the actual α remaining the same [3]. Therefore, less lift is produced when flow separation occurs. As α gets larger, the flow separation also gets larger, hence why the rate at which the C_L is increasing, decreases for morphing airfoils.

In comparison, conventional airfoils generally have a linear trend throughout all α . This is likely due to the slotted flaps, which introduces faster moving air from below into the slowing boundary layer of the upper airfoil surface. This ensures that the flow does not separate, hence keeping the pressure gradient, and pressure difference between the bottom and top surface, producing lift.

Table 1: Percent Error in Matching Cases at $\alpha = 0^\circ$.

Morphing airfoil camber rate (%)	Conventional airfoil flap deflection angle ($^\circ$)	Percent error in matching case (%)
0	-0.25	39.7
2	2.5	5.83
4	5.75	1.83
6	9.5	2.13
8	12.75	0.32

Figure 5: CFD Data for C_d vs α for Various Morphing Rates and α at $Re = 6$ million.



As can be observed in Figure 5, the C_d are generally significantly higher for conventional airfoils than for morphing airfoils. The higher the camber rate is, the higher the initial gap (at $\alpha = 0^\circ$) in the C_d between the cambered airfoils and the conventional airfoils is. The higher C_d for the conventional airfoils are likely due to the slotted flaps. Although the slotted flaps help delay flow separation which delays the onset of pressure drag, they do this by making the boundary layer turbulent, which increases the skin friction coefficient of the airfoil, thus creating additional skin-friction drag. At the same time, the C_d for the morphing airfoils is low because at $\alpha = 0^\circ$,

flow separation has not occurred yet, hence there is little pressure drag contribution to the C_d of the morphing airfoils.

However, it can also be seen that at high α , the C_d of the morphing airfoils surpass the C_d of the conventional airfoils, which can be attributed to the greater flow separation in the morphing airfoils causing the pressure drag created by them to be greater than the skin friction drag created by the conventional airfoils, hence making the C_d of the morphing airfoils larger than conventional airfoils. The slotted flaps in the conventional airfoils help delay flow separation, thus the effect of pressure drag is not as noticeable in the C_d of the conventional airfoils.

Figure 6: CFD Data for L/D ratio of morphing airfoils vs conventional airfoils with increasing α at $Re = 6$ million.

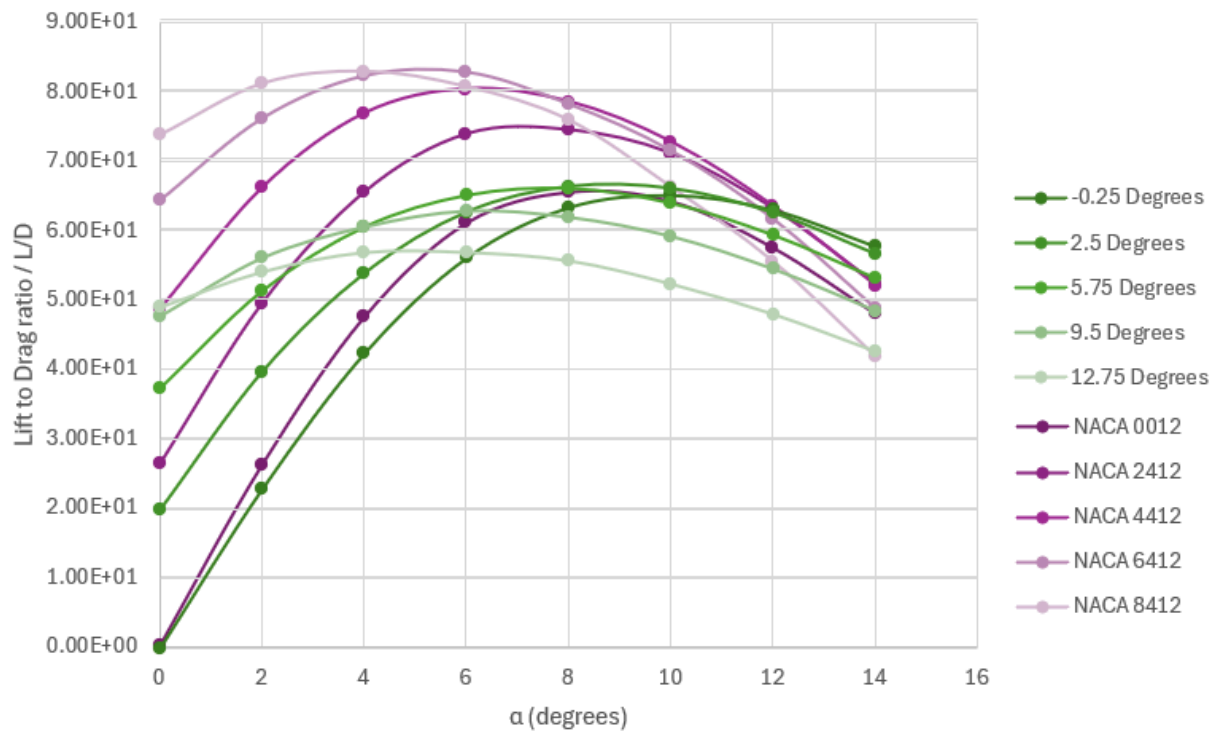
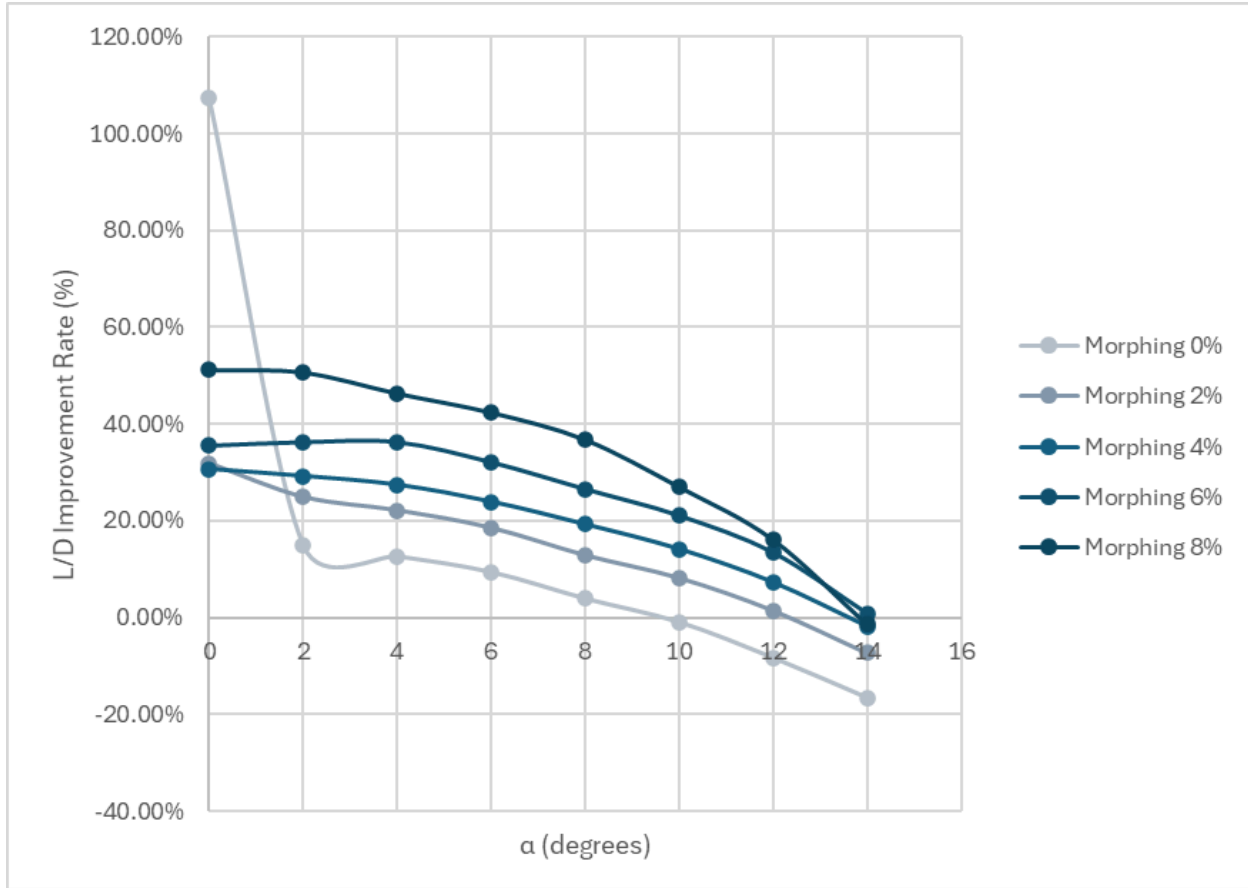


Figure 7: CFD Data for L/D Ratio Improvement Rate of Morphing vs Conventional Airfoils with Increasing α at $Re = 6$ million.



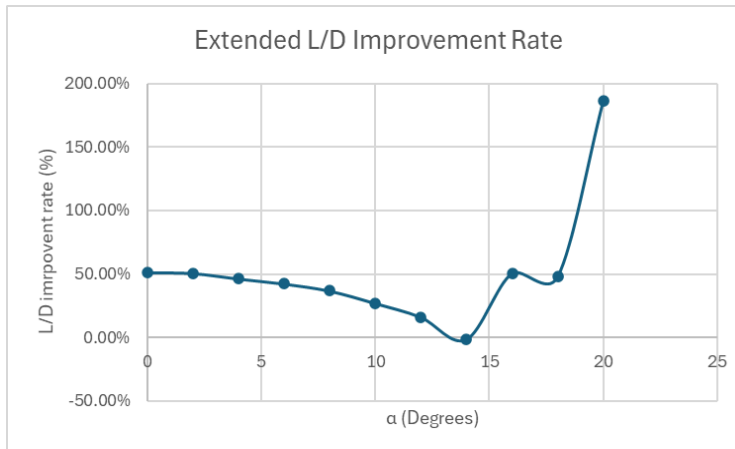
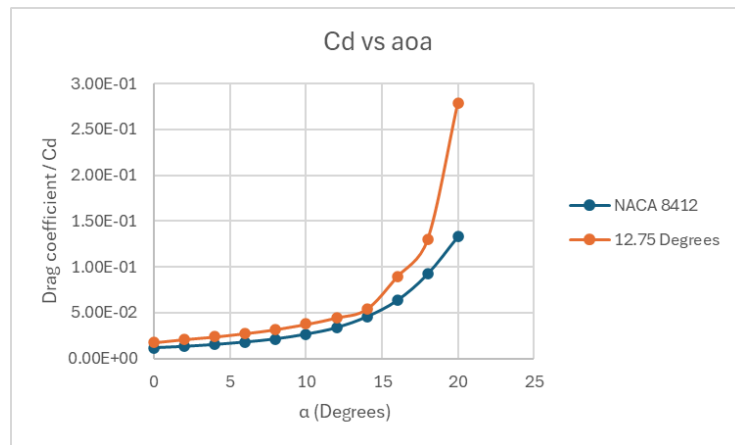
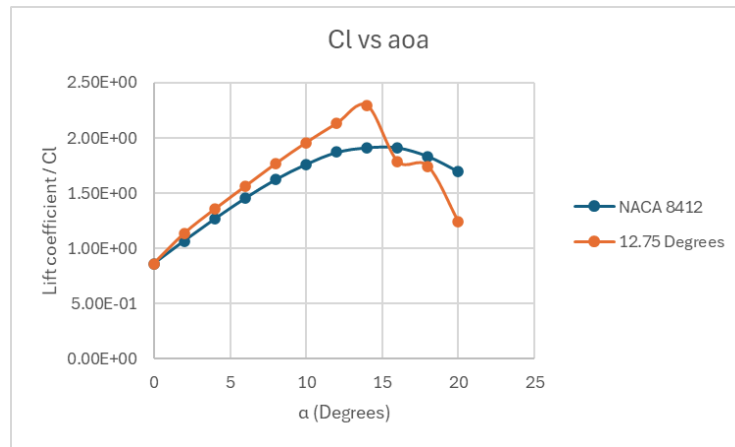
The L/D ratio is a figure that is commonly used when comparing the aerodynamic efficiency of aircrafts. Generally, the higher the L/D ratio, the more efficient an aircraft. This is why the L/D ratio is often the figure that one wants to optimize for when designing an aircraft. Equation X relates the C_L and C_d to the L/D ratio.

$$\frac{L}{D} = \frac{C_L}{C_D} \quad (9)$$

Figures 6 and 7 are plots of the L/D ratios of morphing airfoils vs conventional airfoils. Figure 6 plots the L/D ratio against α while Figure 7 plots how much greater, in percent, the L/D ratio is for morphing airfoils compared to conventional airfoils. Looking at Figure 7, we can observe that at $\alpha = 0^\circ$, the L/D improvement rate ranges between 30% - 50% (ignoring the outlier). The trend appears to be that the greater the camber rate, the greater the L/D improvement rate. This is likely due to the aforementioned reasons of a lack of flow separation at $\alpha = 0^\circ$ causing lower drag and higher lift for the morphing airfoils, hence increasing the L/D ratio.

However, it can also be observed that the L/D improvement rate decreases with α and becomes negative at the highest α . This can again be attributed to the flow separation that occurs for morphing airfoils at higher α , causing lower C_L and higher C_d . Despite this, overall the average L/D improvement was 19.7% across all airfoils and α , ignoring the outlier at the beginning.

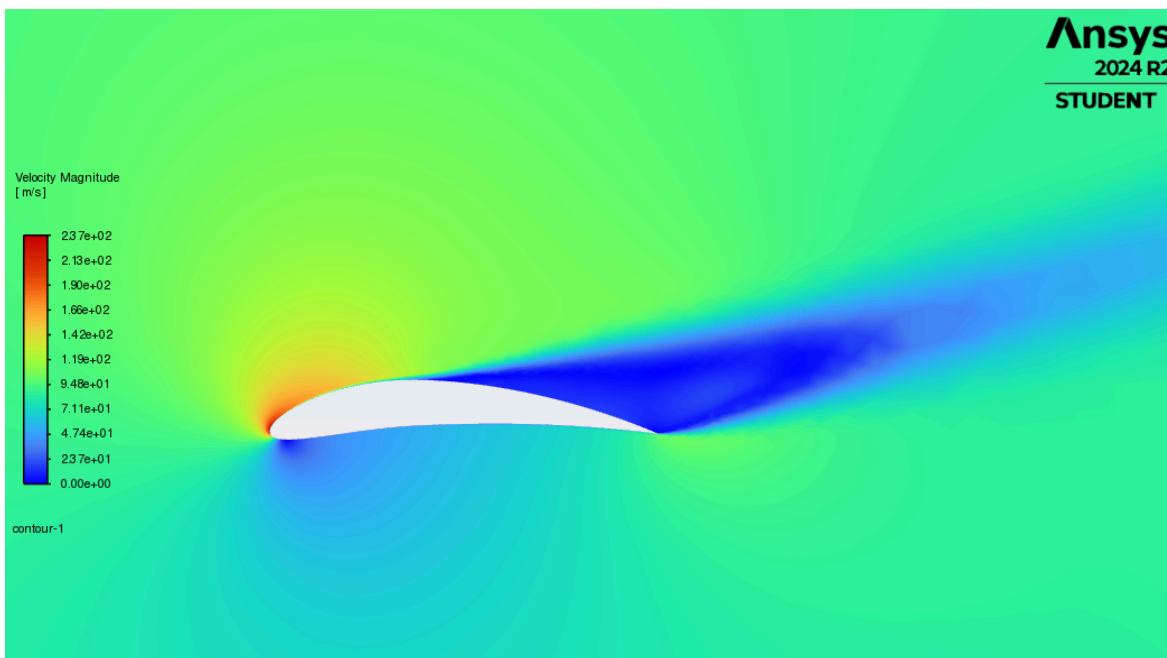
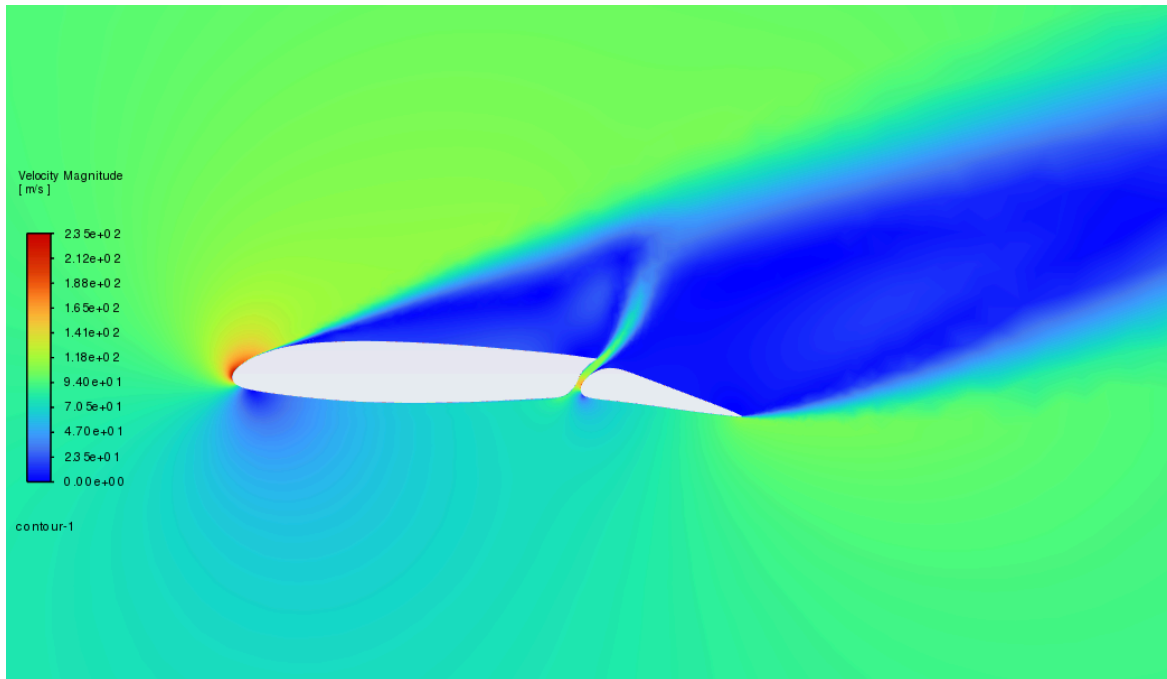
Figure 8a, 8b, 8c: C_L , C_d , L/D Improvement Rate VS α of NACA 8412 vs 12.75° Flap Deflection airfoils.



In Majid et al's paper, it was noted that morphing airfoils tended to have greater L/D ratios throughout all α when compared to conventional airfoils with plain flaps [2]. As this experiment provided differing results, further trials at higher α were conducted for the 8% camber rate airfoil and its matching case. This was done because the Reynolds number used in Majid et al's paper was comparatively very low, around 76,000, while the Reynolds number used in this experiment was 6 million [2]. A greater Reynolds number means a greater freestream velocity, hence also a faster moving boundary layer. As a result of a faster moving boundary layer, flow separation is delayed as it takes even greater adverse pressure gradients to slow the boundary layer to a stop and achieve flow separation. Therefore, in order to test the effects of greater adverse pressure gradients, more trials at greater α were conducted, as a greater α translates to a greater adverse pressure gradient.

The results confirm that more testing was required. At greater than $\alpha = 16^\circ$, the C_L for the conventional airfoil sharply decreases, while the C_d sharply increases. These sharp changes indicate that the airfoil has stalled, which means that the flow has suddenly separated from the airfoil. On the other hand, the C_L for the morphing airfoil gradually decreases, while the C_d gradually increases. Figure 9a and 9b illustrate this on a velocity contour diagram, showing flow separation occurring on both airfoils, with a larger wake on the conventional airfoil compared to the morphing airfoil. If we observe the L/D improvement rate, we see that the L/D improvement decreases till $\alpha = 14^\circ$, after which it suddenly increases again, until at $\alpha = 20^\circ$, it is 186%.

Figure 9a, 9b: Velocity contours comparing equivalent conventional (12.75° flap deflection) and morphing (NACA 8412) airfoil configurations at $\alpha = 20^\circ$, showing a larger wake region for the conventional airfoil.



5: Conclusion

5.1 Limitations of Approach

Numerical methods of solving, as used in this paper, are ultimately approximations and will not be completely accurate. In fact, the numerical method used in this study (steady RANS turbulence models) is not the most accurate one to choose from. There exist other numerical methods such as Unsteady Reynolds Averaged Navier-Stokes (URANS), Large Eddy Simulations (LES) and Direct Numerical Solution (DNS) which make fewer generalizations, and thus are likely to be more accurate [9]. However, these methods are far more computationally expensive, which is why they were not chosen for this study. Furthermore, this study is limited by its use of two dimensional, steady RANS simulations. Three dimensional effects such as induced drag, tip vortices, and spanwise flow are not accounted for. Additionally, steady RANS turbulence models may not accurately predict highly unsteady separated flows near stall. As a result, the results are intended to indicate relative performance trends rather than definitive predictions.

5.2 Efforts to Reduce Error

Before starting the data collection, an experimental validation case study was conducted to confirm the coherence of CFD data with experimental data. The results showed that the CFD data was within 5.21% percent error of the experimental data, which was considered acceptable. Next, the refinement of the mesh was also considered. In Majid et al's paper, there was no

significant variation in the C_d when the number of elements was changed from 100,000 elements to 7 million elements [2]. Since their methodology is similar to this paper, a mesh with close to 100,000 elements was generated for this study. Lastly, increments of 0.25° in the flap deflection angle were used to find matching cases. Finer increments could have been used for more perfect matches, but the average error ignoring the outlier was 2.53%, hence in order to save time it was considered acceptable.

5.3 Summary of results

The study aimed to answer the question: to what extent can the adoption of morphing camber wings help reduce parasitic drag and improve the stall characteristics of aircraft compared to conventional wings with flaps? A reduction in parasitic drag can be achieved by either a reduction in pressure drag or skin friction drag. We saw from the experiments that morphing airfoils tend to have higher pressure drag than conventional airfoils until a critical α , such as 16° for the NACA 8412 airfoil and its matching case, after which the pressure drag also increases for conventional airfoils. In terms of skin friction drag, the slotted flaps in conventional airfoils cause the boundary layer to become turbulent, which increases the skin friction drag. This is why the conventional wings have a higher C_d at most lower α . The main motive behind why the research question aimed to find out if there was a reduction in parasitic drag was in order to determine if the aerodynamic efficiency increased. This is best represented by the L/D ratio metric. As the experiment revealed, there was an average improvement of 19.7% in the L/D ratio of a camber morphing airfoil compared to a conventional airfoil with slotted flaps at α between (0° - 14°). Additional testing of the NACA 8412 airfoil and its matching case at α between (16° -

20°) revealed that conventional flaps eventually stall dramatically, and there is a significant decrease in the L/D ratio. If further α were accounted for, the average improvement in the L/D ratio would likely be higher. Other than differences in aerodynamic efficiency, trends in how the two types of airfoils perform over various α were also found. The further testing revealed that the conventional flaps eventually stall suddenly, which is less desirable than a more gradual stall as seen with the morphing airfoils. In the future, three dimensional simulations and experimental validation should be conducted to further assess the transferability of these findings to real-world wing configurations.

References

- [1] T. Majid and B. W. Jo, “Status and Challenges on Design and Implementation of Camber Morphing Mechanisms,” *International Journal of Aerospace Engineering*, vol. 2021, pp. 1–14, Dec. 2021, doi: <https://doi.org/10.1155/2021/6399937>.
- [2] T. Majid and B. W. Jo, “Comparative Aerodynamic Performance Analysis of Camber Morphing and Conventional Airfoils,” *Applied Sciences*, vol. 11, no. 22, p. 10663, Nov. 2021, doi: <https://doi.org/10.3390/app112210663>.
- [3] J. G. Leishman, “Boundary Layers,” *eaglepubs.erau.edu*, Jan. 2023, doi: <https://doi.org/10.15394/eaglepub.2022.1066.n22>.
- [4] A. Molland and S. Turnock, *Marine Rudders, Hydrofoils and Control Surfaces*. Elsevier, 2022. doi: <https://doi.org/10.1016/c2020-0-01238-7>.
- [5] F. M. White, *Fluid mechanics*, 7th ed. New York, N.Y.: McGraw Hill, 2011.
- [6] F. Roelofs, *Thermal Hydraulics Aspects of Liquid Metal Cooled Nuclear Reactors*. Woodhead Publishing, 2018.
- [7] F. Menter, “Two-Equation Eddy-Viscosity Turbulence Models for Engineering Applications,” *AIAA*, vol. 32, no. 8, Aug. 1994, doi: <https://doi.org/10.2514/3.12149>.
- [8] W. Rodi and N. Fueyo, *Engineering turbulence modelling and experiments 5 : proceedings of the 5th International Symposium on Engineering Turbulence Modelling and Experiments, Mallorca, Spain, 16-18 September, 2002*. Amsterdam ; London: Elsevier, 2002.
- [9] Henk Kaarle Versteeg and Weeratunge Malalasekera, *An introduction to computational fluid dynamics : the finite volume method*. Harlow: Pearson, 1995.
- [10] ANSYS, “ANSYS Fluent Software | CFD Simulation,” *Ansys.com*, 2017. <https://www.ansys.com/products/fluids/ansys-fluent>

- [11] C. Thomas, “NACA2412 Tutorial in ANSYS Fluent (Student Version) - Lift, Drag, Angle of Attack,” *www.youtube.com*, Mar. 06, 2023. <https://www.youtube.com/watch?v=3i9Ryq-m1HA>
- [12] I. H. Abbott and A. E. V. Doenhoff, *THEORY OF WING SECTIONS: Including a Summary of Airfoil Data*. New York: Dover Publications Inc., 1959.
- [13] M. Todorov, “Aerodynamic characteristics of airfoil with single plain flap for light airplane wing,” *INTERNATIONAL CONFERENCE OF SCIENTIFIC PAPER AFASES 2015*, May 2015, doi: <https://doi.org/10.1109/miltechs.2015.7153648>.
- [14] SolidWorks, “3D CAD Design Software,” *Solidworks.com*, 2018. <https://www.solidworks.com/>

6: Appendix

Table 2: Experimental Validation Case Data

Experimental Data		CFD Data	
Angle of Attack (Degrees)	C_L	Angle of Attack (Degrees)	C_L
0	0	0	4.86E-04
0.940006	0.120611	2	2.18E-01
1.96944	0.215533	4	4.37E-01
2.99515	0.34477	6	6.52E-01
3.85131	0.439599	8	8.59E-01
4.87888	0.551678	10	1.06E+00
5.90831	0.6466	12	1.23E+00
7.96346	0.870758	14	1.38E+00
10.1891	1.12074		
11.0471	1.19842		
13.1088	1.36252		

Table 3: C_L Data for Morphing and Conventional Airfoils

Angle of Attack (Degrees)	Morphing					Conventional				
	NACA 0012	NACA 2412	NACA 4412	NACA 6412	NACA 8412	-0.25 Degrees	2.5 Degrees	5.75 Degrees	9.5 Degrees	12.75 Degrees
0	4.86E-04	2.18E-01	4.38E-01	6.52E-01	8.60E-01	2.93E-04	2.05E-01	4.30E-01	6.66E-01	8.62E-01
2	2.18E-01	4.38E-01	6.55E-01	8.66E-01	1.07E+00	2.37E-01	4.36E-01	6.52E-01	8.81E-01	1.14E+00
4	4.37E-01	6.55E-01	8.71E-01	1.08E+00	1.27E+00	4.72E-01	6.61E-01	8.71E-01	1.09E+00	1.36E+00
6	6.52E-01	8.69E-01	1.08E+00	1.28E+00	1.45E+00	7.03E-01	8.83E-01	1.09E+00	1.30E+00	1.56E+00
8	8.59E-01	1.07E+00	1.27E+00	1.46E+00	1.62E+00	9.25E-01	1.10E+00	1.29E+00	1.51E+00	1.77E+00
10	1.06E+00	1.26E+00	1.45E+00	1.63E+00	1.76E+00	1.14E+00	1.31E+00	1.50E+00	1.70E+00	1.96E+00
12	1.23E+00	1.43E+00	1.61E+00	1.76E+00	1.87E+00	1.35E+00	1.51E+00	1.69E+00	1.89E+00	2.13E+00
14	1.38E+00	1.56E+00	1.72E+00	1.84E+00	1.91E+00	1.54E+00	1.69E+00	1.87E+00	2.05E+00	2.29E+00
16					1.91E+00					1.79E+00
18					1.83E+00					1.74E+00
20					1.70E+00					1.24E+00

Table 4: C_D Data for Morphing and Conventional Airfoils

Angle of Attack (Degrees)	Morphing					Conventional				
	NACA 0012	NACA 2412	NACA 4412	NACA 6412	NACA 8412	-0.25 Degrees	2.5 Degrees	5.75 Degrees	9.5 Degrees	12.75 Degrees
0	8.05E-03	8.29E-03	9.00E-03	1.01E-02	1.17E-02	1.01E-02	1.03E-02	1.15E-02	1.40E-02	1.77E-02
2	8.33E-03	8.85E-03	9.91E-03	1.14E-02	1.32E-02	1.04E-02	1.10E-02	1.27E-02	1.58E-02	2.11E-02
4	9.19E-03	9.99E-03	1.13E-02	1.31E-02	1.53E-02	1.12E-02	1.23E-02	1.44E-02	1.80E-02	2.39E-02
6	1.07E-02	1.18E-02	1.34E-02	1.54E-02	1.80E-02	1.26E-02	1.42E-02	1.67E-02	2.08E-02	2.75E-02
8	1.31E-02	1.43E-02	1.62E-02	1.86E-02	2.14E-02	1.47E-02	1.66E-02	1.97E-02	2.44E-02	3.18E-02
10	1.65E-02	1.77E-02	2.00E-02	2.27E-02	2.66E-02	1.76E-02	1.99E-02	2.35E-02	2.88E-02	3.75E-02
12	2.15E-02	2.27E-02	2.53E-02	2.86E-02	3.37E-02	2.16E-02	2.42E-02	2.86E-02	3.48E-02	4.46E-02
14	2.88E-02	2.99E-02	3.31E-02	3.78E-02	4.57E-02	2.69E-02	3.01E-02	3.52E-02	4.26E-02	5.40E-02
16					6.35E-02					1.91E+00
18					9.23E-02					1.83E+00
20					1.33E-01					1.70E+00

Table 5: L/D ratio Data for Morphing and Conventional Airfoils

Angle of Attack (Degrees)	Morphing					Conventional				
	NACA 0012	NACA 2412	NACA 4412	NACA 6412	NACA 8412	-0.25 Degrees	2.5 Degrees	5.75 Degrees	9.5 Degrees	12.75 Degrees
0	6.03E-02	2.63E+01	4.86E+01	6.43E+01	7.37E+01	2.91E-02	1.99E+01	3.72E+01	4.75E+01	4.88E+01
2	2.62E+01	4.95E+01	6.61E+01	7.61E+01	8.11E+01	2.28E+01	3.96E+01	5.12E+01	5.59E+01	5.39E+01
4	4.75E+01	6.55E+01	7.69E+01	8.23E+01	8.29E+01	4.22E+01	5.36E+01	6.03E+01	6.04E+01	5.67E+01
6	6.11E+01	7.39E+01	8.04E+01	8.28E+01	8.07E+01	5.59E+01	6.24E+01	6.48E+01	6.27E+01	5.67E+01
8	6.55E+01	7.45E+01	7.85E+01	7.82E+01	7.59E+01	6.31E+01	6.60E+01	6.58E+01	6.18E+01	5.55E+01
10	6.42E+01	7.11E+01	7.27E+01	7.14E+01	6.62E+01	6.48E+01	6.57E+01	6.37E+01	5.90E+01	5.21E+01
12	5.75E+01	6.32E+01	6.34E+01	6.16E+01	5.55E+01	6.27E+01	6.22E+01	5.91E+01	5.43E+01	4.78E+01
14	4.79E+01	5.23E+01	5.20E+01	4.87E+01	4.19E+01	5.75E+01	5.64E+01	5.30E+01	4.83E+01	4.24E+01
16					3.01E+01					2.00E+01
18					1.99E+01					1.34E+01
20					1.27E+01					4.45E+00

Table 6: L/D Improvement Rate Data of Morphing Airfoils Compared to Conventional Airfoils

Angle of Attack (Degrees)	L/D Improvement Rate				
	Morphing 0%	Morphing 2%	Morphing 4%	Morphing 6%	Morphing 8%
0	1.073796	0.31852	0.306817	0.355206	0.511566
2	0.149017	0.249287	0.292552	0.362262	0.506103
4	0.12637	0.221733	0.27432	0.362565	0.463009
6	0.093631	0.185489	0.239385	0.320942	0.423565
8	0.039012	0.128888	0.192662	0.265479	0.367856
10	-0.00973	0.082014	0.141434	0.210824	0.270376
12	-0.08352	0.014648	0.072714	0.134453	0.161769
14	-0.16607	-0.07299	-0.01826	0.009003	-0.01281