Investigation of a Laminar Airfoil with Flow Control and the Effect of Reynolds Number

Thesis

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Abstract

Wind tunnel tests are performed on a NACA 64-618 airfoil at Reynolds numbers of 6.4x10^4, 1.8x10^5, 1.0x10^6, and 4.0x10^6 in order study several aspects of a laminar airfoil. Studies of flow control, separation bubbles and the effect of Reynolds number are the major topics of this effort. The tools used for investigation are surface pressure measurements, wake surveys, particle image velocimetry, hot-film anemometry, surface-oil flow visualization, and infrared imaging in order to view the problem from many angles. Preliminary testing at a Reynolds number of 64,000 determined that four distinct flow regimes exist with respect to angle of attack: weak laminar separation, moderate laminar separation, laminar separation bubble, and strong leading edge laminar separation. A portion of the study investigates the cause of such dynamic flow physics. Attempts are then made to employ flow control to induce or imitate the laminar separation bubble. By creating the laminar separation bubbles, significant lift increase and drag reduction are realized over a broader range of angles of attack. Normal blowing, suction, and zigzag tape are used, which are all well-characterized devices and have the potential to enhance lift and reduce drag. Lift is increased significantly and separation is delayed in three of the four regions as a result of control, where the region of no change is when the laminar separation bubble is already in effect. It is observed that the optimal flow control device changes between regimes because different flow
physics are required to induce a change. Studies of Reynolds number scaling found that the lift increased and drag decreased as Reynolds number increased. It is important to note that the laminar separation bubble becomes naturally effective at most angles of attack by a Reynolds number of 180,000. Therefore, the value of flow control diminishes except in regions where strong leading edge separation is the limiting element of the airfoil. This research suggests that the laminar airfoil can be controlled in an energy efficient manner such that high performance is gained across all flight regimes with straightforward actuation.
Dedication

Dedicated to my family
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# Nomenclature

\( BR \) = blowing ratio, jet velocity

normalized by boundary layer edge velocity (Equation 12)

\( C_L \) = lift coefficient (Equation 4)

\( C_D \) = drag coefficient (Equation 5)

\( C_P \) = pressure coefficient

(Equation 2 and Equation 3)

\( D \) = drag (Equation 10)

\( F \) = reduced frequency \( \equiv (f \delta) / U_\infty \)

\( FM \) = figure of merit (Equation 11)

\( L \) = lift (Equation 9)

\( L_S \) = length of suction surface

\( M \) = Mach number

\( P \) = pressure; power

\( Re \) = Reynolds number \( \equiv \rho U_\infty c/\mu \)

\( S \) = airfoil area

\( Tu \) = turbulence intensity \( \equiv U_{rms}/U_\infty \)

\( U \) = velocity; uncertainty

\( c \) = airfoil chord

\( c_{a,n} \) = axial or normal force coefficient

\( d \) = diameter, flow control hole

\( f \) = oscillation frequency, Hz

\( s \) = spacing, flow control hole

\( x \) = chordwise distance

\( y \) = wall normal distance

\( z \) = spanwise distance

\( \alpha \) = angle of attack

\( \infty \) = freestream

\( avg \) = average

\( l \) = local

\( rms \) = root-mean-square

\( s \) = static

\( t \) = total

\( w \) = wake
Chapter 1: Introduction

1.1 Introduction

Most current commercial and military aircraft are equipped with airfoils designed to operate with turbulent flow across the airfoil surface due to its boundary layer control feasibility. Airfoils under turbulent flow are highly resistant to separation and are minimally affected by disturbances in the flow because the flow is dominated by inertial forces. A downside to utilizing turbulent flow is that it is unsteady and has a large velocity gradient near the wall resulting in increased skin friction and viscous drag. Because of these downsides, there has been motivation for a primarily laminar flow airfoil due to the reduced skin friction associated with smooth laminar flow. Reduced skin friction decreases drag therefore lowering the required fuel consumption, a critical factor today with the increasing fuel costs and the pressure to reduce environmental impact.\textsuperscript{1} A downside to laminar flow is that it is highly susceptible to boundary layer separation from a strong adverse pressure gradient (e.g. high angle of attack) or transition due to a discontinuity on the surface (e.g. bug deposits or rivets).

The potential for laminar boundary layer separation during a maneuver is highly dependent on the Reynolds number ($Re$) regime in which the aircraft is operating.
Aircraft operating at high Reynolds numbers \( (Re > 1.0 \times 10^6) \), as a majority of the current generation of aircraft are with their high cruising velocities and large chord lengths, use the high momentum and low viscosity flow associated with high Reynolds numbers to help guide the boundary layer further downstream preventing separation from occurring until near the trailing edge. Because the Reynolds numbers are high, less viscous damping exists which enables the growth of instabilities to generate the structures, turbulence and unsteadiness responsible for entraining freestream flow to improve the character of the boundary layer which becomes more resistant to separation.

Several cases arise where a laminar airfoil is introduced into the low Reynolds number flight regime (i.e. takeoff and landing, unmanned and micro air vehicles, wind turbines, sailplanes, and reduced-scale testing). Within the low Reynolds number regime, the effects of viscosity are stronger which suppresses flow and boundary layer instabilities, thus enabling an smooth boundary layer that is extremely susceptible to separation. This causes a multitude of aerodynamic phenomena that necessitate a detailed investigation. Flight in the low Reynolds number regime for a laminar airfoil can lead to premature separation and a significant reduction in the lift-to-drag ratio caused by the inability of the highly laminar boundary layer to follow the steep adverse pressure gradients allowed in more stable boundary layers existent in higher Reynolds number applications.

The flow structures involving separation and control of the boundary layer, especially for laminar airfoils, are not well understood and have become a particularly important field of interest over the past few decades. Many researchers have studied laminar
airfoils, separation bubbles, flow control and Reynolds number regimes on an individual basis, but there is a lack of a comprehensive study of all four together. This thesis attempts to combine the effects of all four. The best method to achieve a comprehensive study is to select a single laminar airfoil, and study the airfoil performance and its dependency on the development of the separation bubble and the physics of control, while varying angle of attack and Reynolds number. The motivation for this research is to reduce the environmental impact of aircraft through reduced fuel consumption which is directly related to the use of a laminar airfoil, where they should ideally be applied to both high and low Reynolds number flows in a cost-effective, optimized manner. To do this, laminar airfoil physics and control must be well understood.

1.2 Objectives

The objective of this thesis is to study laminar airfoils through a wide range of Reynolds numbers to determine a simple solution to increase the performance and operating range of laminar airfoils for both high and low Reynolds number flight. The objectives necessary to perform this study are to:

1) Perform a detailed literature review on laminar airfoil separation, transition, separation bubbles and flow control at both high and low Reynolds numbers

2) Use both a computational model and two wind tunnels to investigate the baseline flow characteristic and airfoil performance of laminar airfoils through a range of Reynolds numbers
3) Introduce different styles of open-loop flow control to the airfoil such that performance is increased significantly with a small amount of required input at both high and low Reynolds numbers

4) Determine the effect of Reynolds number scaling on a laminar airfoil with flow control, where the flow control is also scaled appropriately

1.3 Literature Review

1.3.1 Separation

Separation of flow from the surface of an airfoil is a viscous aerodynamic phenomenon that often limits the angle of attack range of an aircraft due to decreased lift and increased pressure drag. High angles of attack provide the aircraft with significant performance and maneuverability improvement from the increased lift enabling it to make sharp turns and climb quickly, which are attributes important to both civilian and military aircraft. The effort of much airfoil research has gone into extending the angle of attack range by suppressing separation so that pilots have greater safety margins or can push their aircraft to its structural limit in times of dire need.

In order to understand a separated airfoil, the flow dynamics of the inviscid region (i.e. not affected by viscosity) of the upper surface of an attached airfoil must first be examined, as in Figure 1. Point A denotes the stagnation point, where the static pressure equals the total pressure (i.e. Bernoulli’s Equation, Equation 1\(^2\)) and the dynamic pressure equals zero (i.e. the velocity equals zero).
As the flow moves along the upper surface of the airfoil to point B, the flow in the
inviscid region is accelerating. The inviscid acceleration is apparent by the increased
density of streamlines in Figure 1, to maintain continuity of mass through a constricting
area between the airfoil surface and freestream flow. At point B, the dynamic pressure
has risen quickly with a maximum at point C. Because the total pressure at this point has
not changed since no work has been done on the flow, the static pressure at this point
must decrease to satisfy Equation 1. The low static pressure is what provides the suction
force for lift. The fact that the flow area is constantly decreasing from point A to C sets
up a favorable (decreasing) pressure gradient. Since a flow moves naturally from high to
low pressure, it will not separate.

Along with a favorable (decreasing) streamwise pressure gradient up to this point,
there is also a surface-normal pressure gradient which provides the streamline curvature
and allows the flow to remain attached. The pressure starts high in the upper freestream
and decreases towards the surface of the airfoil. The higher upper freestream pressure
provides the centripetal force on the flow to keep it on the surface. Thus, static pressure
is the lowest at the airfoil surface.
Figure 1. Smoke visualization of an airfoil at 0° angle of attack (taken from Munson et al.\textsuperscript{3})

As the flow moves down the airfoil surface beyond point C, the inviscid region begins to slow down from diffusion resulting in a loss of dynamic pressure. Again to satisfy Equation 1, the static pressure must increase, creating an adverse streamwise pressure gradient. Because the pressure is increasing downstream, the surface-normal pressure gradient becomes weaker thus minimizing the streamline curvature.

When viscosity is involved, a boundary layer forms and separation can transpire in the adverse pressure gradient. Within the boundary layer, the total pressure is reduced from viscosity and the static pressure is equal to the inviscid static pressure because the boundary layer is thin, resulting in a reduced dynamic pressure in the boundary layer. If the adverse pressure gradient is great enough, a point is reached when the static pressure becomes so large that the dynamic pressure drops to zero and there is no longer velocity in the boundary layer. Here, the static pressure can no longer increase in the boundary layer to maintain diffusion so it separates from the airfoil to stop further increase in streamwise pressure. Separation from a moderate angle of attack can be seen at point D of Figure 2.
Figure 2. Smoke visualization of an airfoil at 5° angle of attack (taken from Munson et al.3)

Downstream of point D, the airfoil is separated creating a recirculation region which has a high pressure that decreases lift and increases profile drag significantly. Airfoils that have a steep pressure gradient from large diffusion will suffer from separation and reduced performance unless the boundary layer velocity is increased, the adverse pressure gradient is suppressed, or total pressure loss is mitigated.

Boundary layer velocity profiles of a separated airfoil similar to Figure 2 are presented in Figure 3, where the locations A-E are comparable between the two figures. Point A, near the leading edge, has the fullest profile (i.e. the highest relative velocities near the wall) and is not susceptible to separation. However, as the boundary layer moves downstream from point A to point D, it is losing momentum from viscosity and the diffusion-imposed adverse pressure gradient, thus the velocity profile becomes distorted. At point D, it is apparent that the flow is nearly separated because the velocity profile indicates low velocities near the wall with a small gradient normal to the surface. The velocity profile at point E indicates separated flow because there is reverse flow in the velocity profile, which is only possible in a flow that is separated from the airfoil.
It is clear that the separation location is highly dependent upon the angle of attack and airfoil shape. Separation is also dependent on the Reynolds number because of its intimate relationship with total pressure loss and boundary layer thickness from viscosity, and its effect on instabilities and the location of transition. Subtle changes in the camber, location of the maximum thickness and induced flow physics can have a dramatic effect on the location of separation and determine its capability to perform well under adverse conditions.

1.3.2 Transition

At the leading edge of an airfoil, the boundary layer is thin, smooth and laminar. Inherently, disturbances exist in the boundary layer due to wind tunnel imperfections and
airfoil manufacturing tolerances. The propagation of these disturbances depends upon the Reynolds number and pressure gradient present.

Within low Reynolds numbers flow, the viscous forces dominate the inertial forces, so viscosity has the ability to dampen and even eliminate these disturbances. At high Reynolds numbers, the effect of viscosity is less apparent, which allows an increase in the number of small scale structures (because of the decrease in the Kolmogorov length scale) that have the potential to amplify the disturbances. The most general method of increasing Reynolds number is by increasing freestream velocity, increasing airfoil chord length, or increasing distance downstream of the leading edge. Therefore, disturbances are most likely to become amplified far downstream of the leading edge or at higher freestream velocities.

Pressure gradients also play a large role in effecting disturbances. Favorable pressure gradients mitigate the disturbances because as the flow velocity increases, the size of the disturbance stays the same. This reduces the effect of the disturbances compared to the rest of the flow. Likewise, disturbances are amplified in adverse pressure gradients because their size in the mean flow becomes larger.

These disturbances, when amplified, go through a particular process to transition the flow from laminar to turbulent, which is described in Figure 4. The following description is for natural transition, as opposed to bypass transition.\(^4\) As mentioned previously, boundary layers begin as laminar. Then disturbances are amplified, which are typically Tollmein-Schlichting waves for the case of a boundary layer. As these disturbances grow, they begin to generate span- and stream-wise vortices along with other three-
dimensional structures. These structures, although unsteady, are typically beneficial for separation resistance because they bring high momentum freestream flow into the boundary layer and move slow moving flow out which creates a fuller, more stable boundary layer velocity profile. Downstream, these vortical structures breakdown into small, random eddies because of viscosity and insufficient momentum to maintain them. As they break down, small areas of turbulence are generated, known as turbulent spots. Because turbulence dominates over laminar flow, the turbulence propagates outward until the entire flow downstream is turbulent. This entire process is known as transition, where the flow before is fully laminar and after is fully turbulent. Turbulence causes an increase in viscous drag from the higher shear stress at the wall and turbulence energy.

Figure 4. Boundary layer transition on a flat plate (taken from Schlichting et al.)
1.3.3 Laminar Airfoils

1.3.3.1 Design Concept

Because the interest in laminar flow airfoils in the aviation industry has fluctuated with oil prices, the military and airline industry have yet to make a full-fledged leap into the field of laminar flow airfoils because they have seen no long-term need and the cost to make them reliable and effective is very high. A majority of those who are interested in laminar flow airfoils are sailplane and wind turbine designers. Design engineers have made the effort to research and manufacture laminar flow airfoils for such applications, as well as other aerodynamic applications such as aircraft.\(^1\) Because they have spent significant time on the geometrical design of the laminar airfoil, there is little design remaining to improve their performance without a major change. The research now lies in making the laminar airfoil applicable under a wider range of conditions (i.e. Reynolds numbers and angles of attack). This is done by flow control. Now, more than ever, is the time for the aviation industry to get into the laminar airfoil business due to consistently high oil prices and the drive to reduce emissions.

The concept of a laminar airfoil is to create and maintain laminar flow across a majority of the airfoil. The laminar airfoil design was originated by Jacobs around 1935 with the National Advisory Committee for Aeronautics and had its first major implementation on the P-51 Mustang, which showed significant performance increases.\(^6\) The benefit of laminar flow on an airfoil is that it has low shear stress at the wall, because it has a shallow velocity gradient, which relates directly to drag reduction. Reducing drag means the ability to fly with less fuel. For example, one estimate predicted that if the
Boeing 747 could maintain laminar flow on the entire surface of the main wing, 25% of the fuel consumption could be saved equaling 40 tons of fuel per flight.\textsuperscript{7}

In order to achieve a significant amount of laminar flow on an airfoil, long runs of a favorable pressure gradient (i.e. accelerating flow) must be created, which suppress the growth of two-dimensional Tollmien-Schlichting waves in the boundary layer.\textsuperscript{8} In order to physically achieve a long run of favorable pressure gradient, the location of maximum airfoil thickness must be placed near or aft of mid-chord.

The downside of laminar airfoils is two-fold. First, they must be kept extremely smooth so as not to induce any additional instability in the flow which can create turbulence and defeat the purpose of a laminar flow airfoil. This means significant maintenance to polish the wings to remove seams and rivets, and keep them free of contaminants like bug deposits and ice buildup. The laminar airfoils would also be required to fly in smooth, quiet and non-turbulent environments, and keep them from being damaged from debris during takeoff. All of these produce the same results, a transitioned wing which can lead to major safety, reliability and cost issues. Some of these issues are addressed more in depth by Holmes et al.\textsuperscript{8}

The second, and more significant issue, is that laminar flow is more susceptible to separation than turbulent flow. The turbulent boundary layer has more turbulent kinetic energy and momentum associated with it than the laminar case because there is substantial energy associated with the random, swirling motion of the velocity (eddies) that induces entrainment and creates a fuller boundary layer velocity profile. This substantial energy and fuller velocity profile allow the turbulent boundary layer to flow
farther against the adverse pressure gradient before it separates than the laminar boundary layer.

Reynolds number effects are important for laminar airfoils. Seen in Figure 5 at Reynolds numbers above 100,000, the laminar (smooth) airfoil outperforms the turbulent (rough) airfoil, showing the benefit of the laminar airfoil. However, at Reynolds numbers less than 100,000 the laminar airfoil has a significant decrease in performance. This is because disturbances are easily amplified at high Reynolds numbers and structures are less likely to be dissipated by viscosity, whereas at low Reynolds numbers it is more difficult to transition to prevent separation.

![Graph showing performance comparison between laminar and turbulent airfoils](image)

*Figure 5. Comparison of performance between laminar and turbulent airfoils (taken from McMasters and Henderson)*

### 1.3.3.2 Low Reynolds Numbers

The low Reynolds number regime is an important regime to investigate because of the numerous applications and issues associated with it. There are two major conditions
when low Reynolds numbers become a part of the flight plan. The first is when laminar airfoils operating in the high Reynolds number regime ($Re > 1.0 \times 10^6$) enter low Reynolds numbers during takeoff and landing. This is a key to unlocking the future of laminar airfoils, even if the cruise conditions are at high Reynolds numbers. The second condition applies to all small and slow flying wings including: micro air vehicles, unmanned air vehicles, wind turbines, and reduced-scale modeling. Because both conditions use the same laminar airfoil and same regime they will be studied as one.

Several researchers have studied the performance and flow structures of laminar airfoils subjected to a low Reynolds number flow to look deeper into the mechanics of separation and control. Mack et al.\textsuperscript{10} performed an experimental study on a NACA 64\textsubscript{3}-618 airfoil to investigate boundary layer separation and potential applications for control at low Reynolds numbers ($Re \approx 64,000$). Mack et al. showed that at low Reynolds numbers the airfoil was performing below the designed high Reynolds number lift coefficient for the mid-range angles of attack due to premature laminar separation. As the angle of attack was increased to $10^\circ$ in the experiments of Mack et al., a sudden increase in lift and decrease in drag was noted. Mueller and Batill\textsuperscript{11} had similar experimental findings on the NACA 66\textsubscript{3}-018 airfoil as well as Brehm et al.\textsuperscript{12} in their computational study on the NACA 64\textsubscript{3}-618 airfoil, both of which are laminar airfoils.

All three researchers suggested that the performance improvement is the result of a natural boundary layer control mechanism that reattaches the separated boundary layer by transitioning from laminar to turbulent flow early in the separated laminar shear layer. The transition is induced by the Tollmien-Schlichting instability that is amplified as the
boundary layer decelerates and separates. This causes the now turbulent shear layer to re-energize due to turbulent shear stresses and entrain freestream flow. The result is that the boundary layer reattaches at the point where the pressure is approximately equal to that of the fully turbulent solution, creating a closed separation bubble. Although the airfoil is no longer predominately laminar, it has increased performance and angle of attack range.

The flow structure of a laminar separation bubble can be seen in Figure 6. The flow separates at S-S’’ as a laminar shear layer, then becomes turbulent at T-T’-T’’ where the flow quickly reattaches at R-R’’. The area under S-T’-R indicates the separated region. The length of the bubble is dependent on the transition process within the shear layer.

Figure 6. Laminar separation bubble (taken from O’Meara et al.)

Mack et al. held that the bubble acts as a natural flow control mechanism by promoting two-dimensional spanwise vortices (i.e. rollers) downstream of R-R’’ that
replace the low momentum boundary layer fluid with higher momentum freestream fluid therefore stabilizing the boundary layer. The spanwise vortices induced by the laminar separation bubble also have the potential to strengthen particular natural flow frequencies depending on the shedding frequencies, which would encourage vortex-merging thus allowing a further increase in wall momentum exchange.\textsuperscript{12} Both Mack \textit{et al.} and Brehm \textit{et al.} concluded that creating or amplifying the two-dimensional spanwise structures is a very efficient method of boundary layer separation control suggesting that the use of flow control to amplify or create these structures is an excellent candidate for resolving the separation problems at low Reynolds numbers.

As the angle of attack is further increased, the separation point will move closer to the leading edge until transition to turbulence within the separated shear layer no longer expands the shear layer enough to reattach it. At this point, a laminar separation bubble can no longer be created and the airfoil performance suffers significantly. Through the experiment, Mack \textit{et al.} also noted that three-dimensional effects are important, especially at higher angles of attack.

Mack \textit{et al.} also noted that hysteresis played a significant role in separation and reattachment implying that a closed-loop flow control system may be exploited to reduce the momentum required to stabilize the flow after the boundary layer reattached. An example of the hysteresis shown by Mack \textit{et al.} is shown in Figure 7. This shows that once a boundary layer has separated, it takes a significant amount of energy to reattach it. It also demonstrates that once a laminar separation bubble has formed, the angle of attack can be reduced slightly and still have some effect.
Figure 7. Hysteresis of NACA 643-618 at $Re = 64,200$ (taken from Mack et al.)

It is important to note that there are two different types of laminar separation bubbles. Häggmark\textsuperscript{14} made the distinction that “long” laminar separation bubbles strongly alter the pressure distribution whereas an airfoil with a “short” bubble still mimics the inviscid distribution and has little loss of performance as a result of the bubble. Häggmark went on further saying that the short bubble is associated with vortex shedding which confirms the suggestion of Mack \textit{et al.} that the laminar separation bubble is associated with spanwise rollers and often unsteady flow. This indicates that the separation bubble found by Mack \textit{et al.} is short. The long bubble is predominantly steady. Laminar separation bubbles have been studied in depth by Gaster\textsuperscript{15} who suggested that the bubble length decreases with increasing Reynolds number and is a function of angle of attack. The decrease in the length of the bubble as Reynolds number increases makes sense physically because at higher Reynolds numbers, transition occurs earlier.
1.3.3.3 High Reynolds Numbers

Literature and testing regarding laminar airfoils at high Reynolds numbers is less interesting due to the fact that the flow phenomena are not as extreme. Laminar airfoils are optimally designed at high Reynolds numbers to have long runs of laminar flows without significant separation to provide high lift with low drag.

Sommers and Tangler\textsuperscript{16} performed high Reynolds number ($=1\times10^6$) studies on the S825 airfoil, designed as an 18\% thick, high-lift, laminar airfoil for use on large wind turbines. It was found that the maximum lift coefficient increases as the Reynolds number increases. The cause of which is likely due to a thinner boundary layer at increasing Reynolds numbers which allows higher velocities over the upper surface of the airfoil. It was also found that the drag coefficient decreases as Reynolds number increases, due to a thinner boundary layer and a smaller separation bubble if a bubble is present. The increasing $C_L/C_D$ as Reynolds number increases approaches a limit because the boundary layer can only become so thin.

Yamauchi and Johnson\textsuperscript{17} studied the effect of Reynolds number and Mach number variability on the maximum lift and minimum drag coefficients on a variety of airfoils as well and confirmed the previous finding that the drag coefficient generally decreases with Reynolds number increase, and also found that the effect is greater for thicker and rougher airfoils. The maximum lift coefficient also increases with Reynolds number increase where airfoil thickness has a larger effect than roughness. Yamauchi and Johnson also noted the importance of Mach number on the effect of Reynolds number and had some evidence that Reynolds number had less of an effect at higher Mach
numbers. They also mentioned that Mach number corrections should not be neglected even at low Reynolds numbers.

Lift curves from Timmer\(^\text{18}\) on the NACA 643-618 laminar airfoil showed that at high Reynolds numbers \((Re = 6 \times 10^6 \text{ and } 9 \times 10^6)\) the pre-stall lift curve does not change significantly with Reynolds number. However, an increase in Reynolds number in the minor separation \((C_L\text{ maximum})\) region resulted in a higher lift coefficient. Timmer also showed a decrease in drag coefficient as Reynolds number increased.

### 1.3.4 Flow Control

#### 1.3.4.1 Background

Flow control has been used since the dawn of time on throwing spears and darts to allow them to fly fast and straight, but it never became a prevalent scientific investigation until Prandtl\(^\text{19}\) developed boundary layer theory in 1904. Since then, flow control has become very common in the aerodynamic community and is the centerpiece for many research and performance improvement investigations. Many methods of flow control can be adapted to a wing, but the best method is dependent on the type of separation, simplicity and feasibility of actuation, and the cost to operate.\(^\text{20}\) The flow control must also not adversely other performance characteristics of the airfoil which is common on passive methods of control.

In general, flow control enhances performance by changing the flow over the airfoil. There are four major separation control methods: transition from a laminar to turbulent boundary layer, removal of low momentum flow from near-wall, momentum addition to
near-wall, and boundary layer mixing.\textsuperscript{21} As mentioned previously, transitioning the flow from laminar to turbulent creates the potential delay of boundary layer separation due to enhanced mixing and turbulent energy from the small eddies. Removing the low momentum fluid of the boundary layer removes the decelerated fluid near the wall preventing it from becoming stagnant and leading to stall. Removing the low momentum fluid from the wall, if via suction, also typically creates a new boundary layer downstream of actuation. Momentum addition to the near-wall region will increase the velocity gradient and accelerate the flow, and can also be used to generate vortices that can pull the high momentum freestream fluid near the wall and sweep away the low momentum fluid. Boundary layer mixing is a combination of momentum addition and removal through the use of two- and three-dimensional structures that transports momentum from the freestream into the boundary layer to energize and thicken the boundary layer, and transport low momentum flow out of the boundary layer, making it less susceptible to separation.

It is the goal of the engineer to find the flow control device that performs the best for the specific application. Many decades of research have proven that these separation flow control methods work, but now reliability must be improved and net gains must be increased.\textsuperscript{22} Flow control can be categorized in many ways, but it is most common to classify the schemes by passive and active flow control. The following section will review some of the control methods for suppression and delay of separation. This literature review is by no means comprehensive; it is merely a taste of the variety of
methods that can be employed along with a brief review of performance enhancements of each technique.

1.3.4.2 Passive Flow Control

Passive flow control requires no auxiliary power, thus it is permanent and there is no technique to control its strength or turn it off. It will change flow characteristics even when control is not needed, sometimes costing performance by reducing lift and creating a drag penalty. A benefit to this method is that it is easy to apply and operate, and it is very cheap. It can sometimes be easily adapted to airfoils already in use.

Boundary layer trips are the most common method of passive control which uses a surface discontinuity to trip the laminar boundary layer to turbulent. Zigzag tape is often used because it creates the most natural boundary layer transition and creates vortices to enhance boundary layer mixing. A turbulent boundary layer, which has more velocity near the airfoil surface, entrains freestream flow due to the eddies it creates which ultimately keeps the boundary layer attached.

Zigzag tape produces small three-dimensional vortex pairs which are created at each tooth of the zigzag tape as seen by Slangen\textsuperscript{23} in his particle image velocimetry study of tripping devices. A sketch of this effect by Slangen is provided in Figure 8. These vortices are small, yet may entrain a small amount of freestream flow. It is a key point to note that these structures are strongly affected by pressure gradients. The structures are typically strengthened and reduced in size by a favorable pressure gradient, yet they grow and breakdown within an adverse pressure gradient due to lack of energy to sustain and
hold them together. The zigzag tape creates turbulence when the vortical structures induced by the disturbances break down.

![Flow over a zigzag strip (from Slangen)](image)

**Figure 8. Flow over a zigzag strip (from Slangen)**

### 1.3.4.3 Active Flow Control

Active flow control requires energy to power the flow control device. A benefit to this technique is that it can be turned on and off, or modified, at will so losses are not induced when the control is unnecessary. This is preferred because the flow can be adapted to the current flight conditions. The most common active flow control techniques are blowing and suction.

Blowing comprises the majority of active flow control endeavors because it is simple to apply and provides excellent results. Complexity of this device can be increased with parameters like skew (angle of jet relative to freestream direction), pitch (angle of jet relative to airfoil surface), amplitude, frequency, location, and hole dimension, all of which can play a critical role in the actuation physics. For the cases studied here, only
simple control with a normal, steady jet is used to reduce the possible variables affecting the flow physics. The flow physics of a normal jet in cross-flow is discussed in the following paragraphs.

For blowing, Gopalan et al.\textsuperscript{24} has demonstrated that two distinctly different flow structures exist in the wake region behind the jet, depending on the jet-to-crossflow velocity ratio. At low jet velocity ratios ($BR < 2$) the vorticity in the forward side of the jet is cancelled due to mixing with the crossflow boundary layer. Stretching of vorticity in the backside of the jet generates a semi-cylindrical vortical layer (“shell”) as seen in Figure 9. This shell extends from the underside of the jet and connects to the wall, enclosing the domain with slow moving reverse flow. Figure 9 illustrates streamlines and normalized mean velocity magnitude for $BR = 0.5$. Within about one hole diameter above the wall the jet has turned into the crossflow direction and begins to come back to the wall. Gopalan et al.\textsuperscript{24} showed that low values of blowing ratio introduce small disturbances and some streamwise vorticity, similar to zigzag tape. He stated that the strength of three-dimensional vortical structures and disturbances increase with blowing ratio.
Figure 9. Streamlines and normalized mean velocity magnitude at $BR = 0.5$, depicting the domain of slow moving reverse flow formed behind the jet (taken from Gopalan et al.\textsuperscript{24})

At high jet-to-crossflow velocity ratios ($BR > 2$) the jet and its vorticity are advected far from the wall.\textsuperscript{24} Figure 10 illustrates streamlines and normalized mean velocity magnitude for $BR = 2.5$. The jet does not turn into the crossflow direction until approximately three hole diameters above the wall. There is no reverse flow; Gopalan \textit{et al.} suggest that the vortical structures behind the jet resemble a Kármán vortex street off of a cylinder which dominates the instantaneous flow structure.

Figure 10. Streamlines and normalized mean velocity magnitude at $BR = 2.5$, depicting the advection of the jet far from the wall (taken from Gopalan \textit{et al.})
Fric and Roshko\textsuperscript{25} elaborate on the four vortical structures that are evident in the near field of a normal jet in cross flow: horseshoe vortices on the wall, jet shear layer vortices at the perimeter of the bending jet, wake vortices extending from the wall to the jet, and a developing counter rotating vortex pair. Figure 11 illustrates these four vortical structures and their relative size. For this study, the counter-rotating vortex pair and the jet shear layer vortices are most important and are evident in both the high and low blowing ratio cases. The shear layer vortices are what provide the spanwise vorticity (i.e. rollers) and the counter-rotating vortex pair produce streamwise vorticity. Both sets of vortices bring in significant freestream velocity depending on the strength of the jet and if the vortices stay on the surface.

\begin{figure}
\centering
\includegraphics[width=\textwidth]{figure11.png}
\caption{Four types of vortical structures associated with the near-field of a jet in cross flow (taken from Fric and Roshko\textsuperscript{25})}
\end{figure}
It has been shown by Kamotani et al.\textsuperscript{26} that if the normal jets are too close ($s < 4-6d$) the initial vortices of the jet are relatively weak (they vanish as smaller spacing approaches a 2-D jet) and the jets interfere with each other’s entrainment of crossflow. This latter effect results in slower decay of the initial upward momentum flux of the jet. Consequently, for close enough spacing the jets maintain upward momentum longer, and experience only limited deflection due to mutual vortex interaction. However, the spacing used in this experiment (10.67$d$) ensures that the jets will not interfere with each other in the near field. At this spacing the normal jet will behave independently, like a single jet in cross flow.

Suction is the dominant method of flow control in maintaining laminar flow because it will assist in the elimination of an adverse pressure gradient and remove disturbances if properly used.\textsuperscript{27} Suction is a simple method of flow control which extracts near-wall momentum. Typically, this is associated with re-laminarization of the flow not transition to turbulence. This method of flow control is often performed through multiple slots or a porous media to affect a large area or produce a gradient. Suction is a difficult mechanism because it does not have directionality, like blowing, and has difficulty inducing particular flow structures. It is also difficult to drive because there are not many devices on an aircraft with low pressure to suck the air, thus an additional device must be brought on to the aircraft. Blowing, however, can use compressor bleed air as the driver.

1.3.4.4 Low Reynolds Numbers Application

Mueller and Batill\textsuperscript{11} investigated the NACA 663-018, an 18% thick laminar airfoil, to study the laminar separation bubble and airfoil characteristics at Reynolds numbers of
40,000, 130,000, and 400,000. At a Reynolds number of 40,000 at low angles of attack the lift coefficient increased slower than expected as the angle of attack increased because although the velocity on the upper surface of the airfoil increased, the laminar boundary layer separated further upstream. At approximately 8° angle of attack, the laminar separation bubble formed and the lift increased significantly. Tests of surface roughness at the leading edge, which is similar to zigzag tape, showed that the laminar separation bubble forms at the same angle of attack as the smooth case. The change in performance that the surface roughness provided was a more rapid increase in lift as angle of attack increased in the laminar separation region, but also caused the maximum lift coefficient to slightly decrease.

For the $Re = 130,000$ case, the smooth airfoil provided unusual behavior where negative lift was produced at low, but positive, angles of attack. The issue was resolved as transition was induced by the surface roughness. Mueller and Batill noted that airfoil behavior at low Reynolds numbers is very sensitive to airfoil shape, especially the radius of the leading edge, so the results may not represent the characteristics of all laminar airfoils in this Reynolds number regime.

At the upper end of the low Reynolds number range was the case at 400,000. From oil flow visualization, Mueller and Batill were able to show that a very short separation bubble exists, which allows the boundary layer to remain attached longer. They also concluded that the laminar separation bubble decreases in length as the Reynolds number increases; the length of the bubble is also affected more significantly by angle of attack at lower Reynolds numbers. The lift curve showed a distribution similar to that of a high
Reynolds number case, as expected. Surface roughness had little effect on the lift, but increased the drag considerably.

Mack et al.\textsuperscript{10} performed experiments to test both active and passive flow control on a NACA 64\textsubscript{3}-618 airfoil at Reynolds number of 64,200 and 137,000. For the passive flow control, zigzag tape (shown in Figure 12) was used to trip the flow to turbulent which was optimally located at 20\% and 30\% chord locations, for the respective Reynolds numbers. The passive trip made slight improvements in lift and drag, but was only beneficial in particular locations. Mack et al. concluded that a passive trip is not practical because the optimal location, just ahead of the laminar separation point, is constantly changing with the conditions of the flow.

![Zigzag tape used by Mack et al.\textsuperscript{10} (dimensions in mm)](image)

Figure 12. Zigzag tape used by Mack et al.\textsuperscript{10} (dimensions in mm)

For active flow control, Mack et al. used a plasma actuator strip fixed at 40\% chord which pointed downstream and induced a small velocity towards the wall. With the actuator reduced frequency optimally set at $F = 4.29$ using a square wave to excite the
most receptive frequencies, the $C_L/C_D$ was increased 325% over the baseline at $8.3^\circ$ angle of attack.

Brehm et al.\textsuperscript{12} studied active flow control on a NACA 643-618 airfoil in the computational domain. Wall-normal, harmonic blowing and suction were used at a Reynolds number of 64,000 and an angle of attack of $8.64^\circ$. The results indicated that placement of the blowing flow control is most beneficial near the leading edge, at 2% chord. To exploit the hydrodynamic instability, they optimally actuated at reduced frequencies of 0.45 for blowing and 0.36 for suction with a fixed blowing ratio of 0.1. For blowing, Brehm et al. was able to decrease the drag by 58% while maintaining the same lift. For suction, a lift increase of 5% was achieved, while drag was reduced by 55%. In conclusion, they determined that suction was a slightly more effective method of flow control.

A study performed by Agarwal et al.\textsuperscript{28} demonstrated the effect of slot blowing actuator location, reduced frequency and velocity ratio on a NACA 0015 airfoil at $Re = 420,000$. In conclusion, it was found that to maximize the delay of a turbulently separated airfoil from approximately $17^\circ$ angle of attack to $22^\circ$ while enhancing lift by 80% and decreasing drag by 10%, a blowing ratio of 1.2 should be used in conjunction with a harmonic reduced frequency of 1.11 at the 10% chord location. No chord locations closer to the leading edge were tested though they may have proven even more beneficial. Agarwal et al. indicated that actuation closer to the leading edge further delayed stall while actuation nearer the trailing edge produced higher overall lift.
coefficient but resulted in a more abrupt stall. It was also concluded that higher reduced frequencies would not generate better performance.

In summary, the most important parameters are reduced frequency, forcing amplitude and chord location. A choice of optimal values among studies is difficult as most researchers do not perform an exhaustive study of a parameter, nor do any of them agree on certain parameters. It can be said with some confidence, however, that the optimal location for flow control for this study is near the leading edge.

1.3.4.5 High Reynolds Numbers Application

Sommers and Tangler\textsuperscript{16} investigated the effect of fixed transition on the S825 airfoil, designed as an 18% thick, high-lift, laminar airfoil for use on large wind turbines. Transition was fixed at 2\% and 5\% chord of the upper and lower surface, respectively, using grit roughness that decreased in height as Reynolds number increased to maintain its ratio to displacement thickness. They found that fixed transition increases the displacement thickness of the boundary layer, which suppresses the effect of airfoil camber and therefore reduces maximum lift coefficient by 2\% at a Reynolds number of $2 \times 10^6$. Sommers and Tangler also go on to discuss that the effect of standard roughness on the full surface of the airfoil, which is common among most airfoils due to manufacturing tolerances and limited surface polishing, has a significant impact on the performance of the airfoil. At a Reynolds number of $2 \times 10^6$, the maximum lift coefficient was decreased by 14\% due to roughness. With respect to increasing Reynolds number, when standard surface roughness is applied, the decrease in lift and increase in drag is
amplified because the effect of Reynolds number is intimately related to boundary layer thickness to roughness height ratio.

Seifert et al.\textsuperscript{29} used a blowing slot located at the 10\% chord location to delay separation due to the shockwave-boundary layer interaction on a NACA 0015 at Mach numbers between 0.28 and 0.55. The wake deficit and unsteadiness were significantly reduced due to the generation of vortices in the separated region. The stall angle of attack and lift were increased and drag was decreased as a result of control. It was also concluded that the Mach number had a significant effect on the location of separation, and as Tilmann\textsuperscript{30} noted, the actuation should be located just upstream of the shock so as not to produce a secondary shock.
Chapter 2: Test Apparatus

2.1 Test Procedure

Recalling that the objectives of this research are to investigate the performance and characteristics of the airfoil with respect to Reynolds number and the application of flow control, it is evident that a wide array of tests must be conducted to gain a solid understanding of the airfoil fundamentals. In order to grasp the distinct characteristics of the airfoil, four Reynolds numbers are tested: 6.40x10^4 (M = 0.02), 1.80x10^5 (M = 0.06), 9.88x10^5 (M = 0.11), and 3.96x10^6 (M = 0.33). A Reynolds number of 64,000 is chosen because it has been studied extensively by several other researchers, so the flow features of the baseline are well understood. A Reynolds number of 180,000 is selected because it is the upper limit of the low speed wind tunnel. The lower limit of the high speed tunnel is \( Re = 9.88x10^5 \). A Reynolds number of 3.96x10^6 was decided upon because it shows scalability differences without significant compressibility effects. The testing includes one computational model, XFLR5, and two different wind tunnels.

At each Reynolds number, batteries of tests are conducted to understand the basic flow. The angle of attack is swept from -3° to 22° to obtain the lift curve slope and the drag polar from surface pressure distributions and wake surveys. From here, regions of
distinctly different flow are identified from the lift curve. Significant effort is placed on studying these regions with the application of flow control. At the control angles at 64,000, flow is investigated thoroughly with surface pressure distributions, surface oil flow visualization, particle image velocimetry, and hot-film probes. At higher angles, little additional investigation occurs at the four control regions due to limitations in the apparatuses.

2.2 XFLR5

The computational model used for this investigation is XFLR5 which is a freeware code under the General Public License developed by Mark Drela and Harold Youngren of the MIT Aeronautics and Astronautics Department. XFLR5 uses XFOIL as the aerodynamic solver with an enhanced user-interface; XFLR5 and XFOIL both provide the same results.\textsuperscript{33} XFOIL is an aerodynamics tool used for the design and analysis of subsonic, isolated airfoils and sailplanes. It is a fast and accurate high-order panel method with a fully coupled viscous-inviscid interaction method. XFOIL is capable of calculating lift and drag, with the inclusion of Reynolds and Mach number variation. The developers stress that the code is reasonable and consistent for model sailplanes, but should not be used for the design of real-sized aircraft. They also do not guarantee the accuracy and robustness of the code.

For the purposes of this project, XFLR5 is used to calculate the lift curve and drag polar for the airfoil under investigation at the scheduled Reynolds and Mach numbers under free transition with a standard $N_{crit}$ value of 9. To capture the physics without
excessive calculation time 150 panels are used which are freely distributed by XFLR5 to their optimal location. It should be noted that these calculations are done two-dimensionally, although three-dimensionality effects are important especially at separation of high angles of attack.\textsuperscript{10}

2.3 Wind Tunnel

2.3.1 Low Reynolds Number

The wind tunnel used for the low Reynolds number testing is the straight section of the open-loop low speed wind tunnel at the Aeronautical and Astronautical Research Laboratory (AARL) of The Ohio State University, shown in Figure 13. This wind tunnel is powered by a 40 HP American Fan Company centrifugal blower, which pushes the air through a series of flow conditioners and enters a rectangular (0.381m x 0.362m) acrylic-walled duct. The wind tunnel is safely capable of operating at chord Reynolds numbers between 0 and 180,000. The selected Reynolds numbers for testing in this tunnel are 64,000 and 180,000. The freestream turbulence level at the two Reynolds numbers is 0.45\% at 64,000 and 0.74\% at 180,000. The airfoil can be mounted both horizontally and vertically in the test section with angle of attack markers between -10° and 25° at every one degree. Because the tunnel walls are acrylic, the test section has clear optical view for the use of particle image velocimetry and flow visualization.
2.3.2 High Reynolds Number

Figure 14 provides an illustration of the transonic wind tunnel used for the high Reynolds number portion of the study which is also located at the AARL. It is an open-loop, blow-down type wind tunnel with a test section height of 559 mm and a width of 152 mm. The upstream high pressure air is driven by two compressed air tanks which are controlled by a Belfield valve. The tank pressure and Belfield valve are used to regulate the test section total pressure which sets the Reynolds number. This wind tunnel is subsonic at the test section, but becomes sonic at a downstream throat preventing the downstream conditions from affecting the test section. The Mach number is varied by adding or removing blockage with choke bars at the throat. In the case of shock formation in the transonic regime, reflected wall shocks are prevented from affecting the airfoil flow characteristics by using perforated plates at the upper and lower surface of the wind tunnel wall.

The range of Mach and Reynolds numbers the transonic tunnel is capable of operating under normal conditions can be seen in Figure 15. The tunnel extends to higher Mach and Reynolds numbers but the extrapolation used to create this plot is not trusted.
beyond a Mach number of 0.55. The blue lines indicate the upper and lower limits of the tunnel which were created with a regression of the collected data points. The red circles indicate the two test points. Testing only occurs at Reynolds numbers of $9.88 \times 10^5$ and $3.96 \times 10^6$ with Mach numbers as low as possible to reduce the effects of compressibility. The freestream turbulence level at these conditions is measured to be approximately 0.2% with a hot-film. The high pressure freestream air provides approximately ten seconds of data acquisitioning time after steady state conditions have been reached. The angle of attack is measured with an inclinometer with an accuracy of 0.011°.

For simplicity, from here forth, Reynolds numbers of $9.88 \times 10^5$ and $3.96 \times 10^6$ will be nominally denoted as 1 million and 4 million, respectively.

Figure 14. Schematic of the high speed wind tunnel (taken from Gompertz et al.$^{34}$)
2.4 Airfoil

Because the flow and performance characteristics of the uncontrolled NACA 643-618 airfoil are well known,\textsuperscript{10,12,31,32} it is used as the airfoil for the following investigation on laminar airfoils. A drawing from XFLR5 of the airfoil is shown in Figure 16. The NACA 643-618 airfoil has a maximum thickness of 18% at the 37.1% chord location. It has a camber of 3.31% with the maximum deviation point at 51.4% chord. The location of the maximum thickness indicates that the airfoil is laminar because it provides a long run of favorable pressure gradient which is necessary to suppress boundary layer instabilities and maintain laminar flow along a majority of the airfoil chord.
2.4.1 Low Reynolds Number

The airfoil tested in the low speed tunnel is fabricated by milling aluminum ribs, connecting them via spars and gluing a thin aluminum sheet over the airfoil. The skeleton of the airfoil without the aluminum surface is shown in Figure 17. The span of the airfoil is 362 mm and the chord is 153 mm. Thirty-eight pressure taps (29 on the upper surface 9 on the lower surface) have been drilled into the middle 1/3 span of the airfoil surface. The chordwise locations of the pressure taps and flow control are shown in Figure 18. The pressure taps are staggered at a 15° angle to avoid interference. The non-uniformity shown between the pressure tap locations is due to geometric constraints from spars.
2.4.2 High Reynolds Number

For application in the high speed wind tunnel, a stronger airfoil has been fabricated to withstand the higher aerodynamic forces to which it would be subjected. The same
NACA 643-618 laminar airfoil shape is used with a 152 mm chord and a 152 mm span, which is made of solid aluminum. Thirty-one surface pressure taps (24 on the upper surface and 7 on the lower surface) have been drilled along the centerline of the airfoil. The chordwise position of the surface pressure taps and flow control is shown in Figure 19. An isometric view of the model is shown in Figure 20 which includes the location of the pressure taps and flow control. The blocks on the end of the airfoil, which have been filleted with the airfoil surface to prevent discontinuity, are used to mount the airfoil in the test section. The tap locations are not staggered as it was deemed unnecessary; some staggering is implemented at the leading edge due to the congestion of taps and flow control in that area.

Figure 19. High speed NACA 643-618 airfoil with tap and control locations
2.5 Flow Control

Three methods of flow control are employed in this study: zigzag tape, suction and blowing through normal holes. These three methods are chosen because they each align with the different types of control mentioned in Section 1.3.4.1. As suggested from previous research studies,\textsuperscript{10,12,28} all three methods are placed at the same location as close to the leading edge as possible. The near leading edge position is constrained by internal airfoil fabrication and is thus located at 5% chord, as seen in Figure 18.

2.5.1 Low Reynolds Number

The zigzag tape, which is used as a method of passive flow control to trip the boundary layer from laminar to turbulent, has a thickness of 0.2 mm, a width of 5.35 mm and a peak-to-peak distance of 2.8 mm. Figure 12 depicts the geometry of the tape; all dimensions are taken from tape used by Mack \textit{et al.}\textsuperscript{10} but scaled with respect to the chord.
The active flow control for both the blowing and suction use the same internal system. The blowing is powered by 100 psi shop air and the suction is driven by an industrial vacuum. Figure 21 shows the ribs and spars, along with the extensive tubing required to get a uniform distribution of flow through the control holes. Externally, round holes were drilled normal to the airfoil surface, which have a diameter of 0.794 mm and a spanwise spacing, $s$, of 8.467 mm ($10.67d$). The diameter and spacing are 0.48%$L_S$ and 5.13%$L_S$, respectively, which compare well to the percentages employed by similar research. The flow control holes run 93% of the airfoil span.

![Figure 21. Infrastructure of active flow control in low speed airfoil](image)

A King Instruments Company rotameter controls the volume flow rate for both blowing and suction at lower Reynolds numbers. For $Re = 64,000$, a rotameter which measures SCFH (standard cubic feet per hour) from 6-60 by increments of two is used. At high Reynolds numbers, GPM (gallons per minute of water) of a separate rotameter is measured from 0.1-1.0 in increments of 0.02. Both rotameters can be set accurately to $\frac{1}{4}$
increments. Both are converted to the unit standard SCFH for air which is used to obtain blowing ratios.

2.5.2 High Reynolds Number

The flow control applied to the high Reynolds number cases use only normal blowing. The infrastructure for the high speed airfoil consists of a cylindrical cavity drilled spanwise through the airfoil at 5% chord. Normal holes with the same dimensions as the low Reynolds number airfoil are drilled through the surface to the cavity. The cavity and normal holes can be viewed in Figure 20. Tubing is brought in to both ends of the cavity to ensure a uniform distribution of blowing. The tubing is attached to the same rotameters and driven by the same high pressure line as used in the low Reynolds number actuation. The impact of the holes is negligible on the baseline flow of the airfoil.

2.6 Data Acquisition

2.6.1 Pressure Measurement

For the Re = 64,000 study, the surface pressure taps are connected to a 0.5 inH₂O (125 Pa) Druck ultra low differential pressure transducer to obtain the static pressure distribution along the airfoil surface. The Druck is model LPM1010 which is unidirectional with an accuracy of ±0.5% after calibration. The sensor has a frequency response of 100 Hz. At Re = 180,000, the pressure differential is too large for the Druck transducer so instead a ±0.5 psi Validyne differential pressure transducer is used. The
P24 Validyne model used has a frequency response of 1000 Hz with an accuracy of ±0.5%.

For the higher Reynolds numbers cases (1 and 4 million), different pressure measurement devices are necessary because the runs are too short to run through each tap manually and the pressures are significantly higher. Two Esterline 9116 Intelligent Pressure Scanners are used to obtain the surface pressure on the airfoil. These scanners are capable of operating up to differential pressures of ±45 psi with a frequency response of 500 Hz and a full scale system accuracy of ±0.05%. Such high performance is obtained through the use of 16 silicon piezoresistive pressure sensors on each scanner.

The total and static freestream pressures are obtained with an upstream Pitot-static probe for the low speed wind tunnel and a stagnation chamber Pitot tube with upper- and lower-wall static taps at the test section for the high speed tunnel.

Wake pressures are acquired by traversing a Kiel probe in the low speed tunnel and a Pitot probe in the high speed tunnel across the wake along the airfoil centerline. A Kiel probe is useful because it is similar to a Pitot tube; however, it has a shroud allowing the probe to accurately obtain pressure at angles slightly offset from the freestream. In the low speed tunnel, the probe is traversed 1.67c downstream of the trailing edge. The high speed probe is also traversed at a downstream location of 1.67c from the trailing edge.

For the lift pressure data in the low speed tunnel, five seconds of data are acquired at a frequency of 100 Hz \((Re = 64,000)\) or 1000 Hz \((Re = 180,000)\) for each pressure tap. The drag data in the low speed tunnel is collected at the same frequencies with an 8 second settling time and 7 second acquisition time.
The high speed data acquisition is more rapid due to limited flow pressure. Data is acquired at 100 Hz for both lift and drag data. Because Reynolds number decreases as the tunnel runs, only the small portion of data where the traverse is in the wake region is being used to simultaneously calculate lift and drag for the same Reynolds number. The length of data acquisition time is dependent upon the width of the wake, but is on average about 1.5 seconds. Therefore each tap has a sample of 150 data points, but each wake point is only one sample, with 150 points in the wake.

2.6.2 Particle Image Velocimetry

In addition, particle image velocimetry is employed for flow visualization purposes, and is utilized primarily to investigate the flow character and locations of separation, transition and reattachment at a given angle of attack. A single, high speed LaVision particle image velocimetry system is used to obtain two-dimensional velocity data. The camera has a resolution of 1376 by 1040 pixels and is oriented to capture flow phenomena running the full length of the chord. Two image fields are taken, one at the leading edge half and the other at trailing edge half, and meshed together to get a high resolution, full field data set. For the present study, three-dimensional effects are neglected and consequently the particle image velocimetry images are secured near 50% span on a flow control hole.

A double pulsed Nd:YAG laser is used to project two consecutive 1mm thick laser sheets (with a 25µs time separation) in the x-y plane into the test section. Olive oil particles with diameters between 1 and 2µm are inserted into the flow as seed. The laser pulses are synchronized with the high-speed camera to capture two exposures of the light
scattered by the olive oil particulate. One thousand image pairs are acquired and processed for each measurement. Two 32 x 32 pixel interrogation windows, each one overlapping another by 50%, are used to compute spatial correlation and output velocity vectors. From the particle image velocimetry analysis, the full-field mean velocity characteristics are obtained and used to diagnose the flowfield.

An image of the particle image velocimetry setup is provided in Figure 22, which shows the laser, camera, seeding, airfoil and the two overlapping image fields.

![Figure 22. Particle image velocimetry setup](image)

### 2.6.3 Surface Oil Flow Visualization

Oil flow visualization is used on the surface of the airfoil using a combination of brake fluid as the viscous transport and blue ink toner as the location identifier. The purpose of oil flow is to demarcate the separation line and possible regions of laminar and turbulent flow using the properties of shear stress on the surface of the airfoil. The
combination is atomized and sprayed evenly on the upper surface of the airfoil, which is then inserted horizontally into the wind tunnel. The flow is then turned on and images are taken of the airfoil once the fluid has reached a steady-state condition. Separated regions, where the shear stress is low (indicated by unmoved oil), and regions of large shear-stress gradients (indicated by oil-accumulation lines) are nicely identified with this approach.

2.6.4 Infrared Imaging

Some infrared imagery is obtained using an Electrophysics Silver 420 shortwave infrared camera. The camera has a sensitivity of 0.02°C, due to cryogenic cooling of its detector, to a temperature of 70K. The camera has a maximum frame rate of 100 Hz with a temperature measurement range of 5 to 1500°C. To capture data, the flow is heated and diverted from the test section until it reaches steady-state of about 50°C while the airfoil remains at room conditions. The flow is then directed down the wind tunnel and the data is acquired. The hottest locations are indicated by red followed by orange, yellow, green and blue.

2.6.5 Hot-film Anemometry

For better turbulence statistics and near-wall data, a single sensor hot-film anemometer is used to obtain mean velocity, turbulence intensity and power spectral density information. The film has a diameter of 50.8 μm and a length of 1.02 mm and is low-pass filtered at 10 kHz to attenuate noise. The hot-film is mounted to a traverse that takes chordwise slices in-line with a midspan flow control hole using a follower device
which allows the hot-film to maintain a constant normal distance from the airfoil surface during the traverse. The chordwise distance between each data point is \( x/c = 0.0262 \) and the wall normal distance is \( y/c = 0.0196 \) (3mm). The slices are always initiated at the leading edge, but do not reach the trailing edge due to follower device constraints. Further, at selected chordwise locations the hot-film is used to acquire wall-normal plunges for boundary layer data. The true wall normal distance is unknown; the nearest wall location is optically determined to be less than 0.1mm. Therefore the nearest wall location, in all analysis, is set to 0.1mm, and all other distances are adjusted relative to that.

### 2.7 Data Reduction

Time-averaged pressure coefficient calculations are made at each of the tap locations where \( C_p \) is defined as in Equation 2 for incompressible flows and in Equation 3 for compressible flows, since Bernoulli no longer applies.\(^{36}\)

\[
C_p = \frac{P_s - P_{s\infty}}{P_{t\infty} - P_{s\infty}} \quad \text{Equation 2}
\]

\[
C_p = \frac{P}{P_{s\infty}} - 1 = \frac{1}{0.5\gamma M_{\infty}^2} \quad \text{Equation 3}
\]
Using the trapezoidal rule to integrate the $C_P$ to obtain the normal ($c_n$) and axial ($c_a$) force coefficients, the two-dimensional lift coefficient, $C_L$, is consequently determined in Equation 4.

$$C_L = c_n \cos(\alpha) - c_a \sin(\alpha) \quad \text{Equation 4}$$

Drag coefficient is calculated by using the momentum equation in the wake, as defined by Equation 5.

$$C_D = \frac{2}{c} \int \frac{\rho_w U_w}{\rho_\infty U_\infty} \left( 1 - \frac{U_w}{U_\infty} \right) dy \quad \text{Equation 5}$$

These lift and drag values are corrected for wind tunnel effects per the equations developed by Barlow et al., which are provided in Appendix A. For the Reynolds number comparison figures only, because the effect of compressibility should be removed for direct comparison, the Prandtl-Glauert compressibility correction is used in reverse such that all values are presented as incompressible. An uncertainty analysis is performed on the lift coefficients which are shown as error bars on the $C_L$ versus $\alpha$ curve. The derivation is provided in Appendix B.

In order to determine airfoil performance, two different measures are used for comparison between baseline and flow control. The first is percent lift increase and percent drag decrease, which are shown in Equation 6 and Equation 7, respectively.
Although these are simple, yet telling performance measures, they provide no information about the effectiveness of actuation or the energy efficiency of the control method. Therefore, the figure of merit defined by Collis et al., presented in Equation 8, is employed. This equation is a ratio of the lift-to-drag ratio of the control case over the baseline case. The control case has an additional term which is the power of actuation ($P$), an effective drag. The lift and drag are multiplied by freestream velocity to obtain their corresponding power so they can be non-dimensionalized with the actuation power. The actuation power is calculated as the pressure differential required to drive the flow times the volume flow rate.

To get the figure of merit in a form that can be dealt with, Equation 9 and Equation 10 are used to convert lift and drag into their corresponding coefficients, which are known along with density, freestream velocity and airfoil area. The resulting figure of merit is provided in Equation 11. A figure of merit greater than one indicates that the power required to introduce the flow control to the airfoil is worth the effort. This is a

\[
\% \text{ Drag Dec} = -\frac{C_{DControl} - C_{DBaseline}}{C_{DBaseline}} \times 100 \quad \text{Equation 6}
\]

\[
\% \text{ Lift Inc} = \frac{C_{LControl} - C_{LBaseline}}{C_{LBaseline}} \times 100 \quad \text{Equation 7}
\]
A convenient method of determining if it is energy efficient to pull power from somewhere else in the aircraft to drive the actuation.

\[
FM = \frac{\left[ \frac{LU_\infty}{D U_\infty + P} \right]_{\text{control}}}{\left[ \frac{L}{D} \right]_{\text{baseline}}} \quad \text{Equation 8}
\]

\[
L = \frac{1}{2} C_L \rho U_\infty^2 S \quad \text{Equation 9}
\]

\[
D = \frac{1}{2} C_D \rho U_\infty^2 S \quad \text{Equation 10}
\]

\[
FM = \frac{\left[ \frac{1}{2} C_L \rho S U_\infty^3 \right]_{\text{control}}}{\left[ \frac{C_L}{C_D} \right]_{\text{baseline}}} \quad \text{Equation 11}
\]
Chapter 3: Results

3.1 Roadmap

The following section contains the results and a discussion of the tests described in Section 2.1. This chapter will have five major sections: one for each of the four Reynolds number cases, and a section to compare them as a whole. Recall the nominal Reynolds numbers are $6.4 \times 10^4$ (Case I), $1.8 \times 10^5$ (Case II), $1.0 \times 10^6$ (Case III) and $4.0 \times 10^6$ (Case IV). Each Reynolds number case is broken down into a study of the baseline characteristics followed by the use of control techniques at various angles of attack. A variety of investigation techniques are used, which was described in Section 2.6, to help determine location of separation, transition, measures of performance and the global flowfield.

Case I beholds the most interesting flow physics and is where a majority of the effort is spent. The angles of interest and control are determined from Case I and are scaled appropriately as the Reynolds number increases through each case. The discussion of the effect of scaling the Reynolds number on the performance of the airfoil and the change in effectiveness of the control is presented at the end of the chapter.

Previously reported data$^{40}$ may show a different angle of attack than what is presented here at $Re = 64,000$ which is the result of a wind tunnel correction. The previous data had
a two degree shift in angle of attack to match the location of the formation of certain phenomena noted by Mack et al.\textsuperscript{10} This, however, did not match the high Reynolds number data acquired here in an independent study, where the angle of attack is well known. The angle shift was reviewed and determined to be incorrect, therefore the data presented in the following is the actual measured angle of attack.

3.2 Case I: $Re = 6.4 \times 10^4$

3.2.1 Baseline

Before evaluating the characteristics of the NACA 64\textsubscript{3}-618 laminar airfoil, efforts are made to compare the baseline lift and drag coefficients of these experiments to the results of XFLR5, Mack et al.\textsuperscript{10} and Brehm et al.\textsuperscript{12} to ensure the experiments performed here are comparable to others’ work. These results are shown in Figure 23 and Figure 24 as the lift coefficient against the angle of attack and drag coefficient, respectively. The presented lift coefficient is calculated by surface pressure measurements and the drag coefficient is calculated by a wake traverse, both discussed in Section 2.6.1. Uncertainty bars have been included with the lift coefficients.
Figure 23. $C_L$ vs. $\alpha$, $Re = 64,000$, baseline comparison of NACA 643-618

Figure 24. $C_L$ vs. $C_D$, $Re = 64,000$, baseline comparison of NACA 643-618
From this brief investigation, it is readily apparent that the airfoil developed for this research effort displays similar lift and drag characteristics to all three comparisons. From $\alpha = -1^\circ$ to $12^\circ$, the experimental data lift coefficient follows the same slope and magnitude as Mack \textit{et al.} The data of Mack \textit{et al.} and Brehm \textit{et al.} jump to higher lift and lower drag values around $\alpha = 10^\circ$ which corresponds to the formation of the laminar separation bubble. The experimental data obtained by this effort does not show the formation of the laminar separation bubble until closer to $14^\circ$ which is the result of slight differences in test facilities and geometry that are exacerbated by the extreme sensitivity of this airfoil at low Reynolds numbers. It is also evident that the airfoil does not have a $C_L$ plateau as Mack \textit{et al.} and Brehm \textit{et al.} indicate, and instead shows a peak and drop-off. Beyond the formation of the laminar separation bubble at $\alpha > 18^\circ$ the lift and drag closely follow the XFLR5 data.

The drag polar shows an interesting trend where the drag coefficient rapidly increases with lift coefficient, which demonstrates the laminar separation point moving forward and the separated region growing. A considerable decrease in drag and increase in lift, corroborated by the data of Mack \textit{et al.}, Brehm \textit{et al.} and XFLR5, occurs which is the region of the laminar separation bubble. Then the laminar separation bubble disappears and the drag falls back onto the curve of laminar separation.

Mack \textit{et al.} noted that the airfoil they developed was slightly modified. The airfoil fabricated for these experiments may have slight differences in leading edge radius and thickness resulting in the formation of the laminar separation at a different angle of attack. Also, it is suspected that the wind tunnel turbulence values differ between these
and Mack et al.’s tests, as well as differences in surface roughness values, which is likely causing the transition point to change. It is known that freestream turbulence, surface roughness and geometry affect the performance of a laminar airfoil, so a detailed investigation of these effects is not performed to confirm the root cause. The main purpose of this comparison is to demonstrate that the airfoil matches its expected performance characteristics and the data to follow are valid.

Investigation of the baseline curve revealed four regions of unique behavior at this Reynolds number, which have been denoted in Figure 25 and will be discussed in the following sections. It is important to note that the endpoints of these regions are not well defined due to the limited resolution of the $C_L$ vs. $\alpha$ plot. The divisions should be regarded as regions of distinctly different character, without a hard cutoff angle.

![Figure 25. Regions of NACA 64\(_{3}-618\) at $Re = 64,000$](image-url)
3.2.1.1 Region I

For $\alpha < 0^\circ$ (Region I), it is expected that the airfoil experiences minor separation on the upper surface of the airfoil, per suggestion by Mack et al. Mack et al. also assert, based on measured surface-pressure distributions and numerical simulations, that the flow is laminar. Because the region is predominately attached and laminar, it is hypothesized that flow control at this angle will do little to improve lift; this will be discussed later. The lowest angle available from the baseline sweep, $\alpha = -1^\circ$, will be used as the Region I angle to be investigated with control and other measurement techniques.

The first method of investigation is to look at the upper surface pressure distribution at $\alpha = -1^\circ$, which is shown in Figure 26, and compare the experimental results to the viscous and inviscid computational results from XFLR5. It is obvious that the pressure distribution does not follow the inviscid XFLR5 prediction; however, the viscous solution matches exactly in character and is approximately close in magnitude. This distribution indicates separation at $x/c \approx 0.5$, by the constant pressure aft of the separation point. Interestingly, the flow separation is not minor, as expected, where instead it separates near the point of maximum thickness where the airfoil surface begins to diverge from the freestream flow. It is sensible that separation occurs at maximum thickness because it does not have enough inertia to follow the curvature and adverse pressure gradient. The experiment and XFLR5 viscous solution also both indicate reattachment of the flow at the trailing edge. This is a phenomenon suggested by Mack et al. to be a trailing edge laminar separation bubble. It can be concluded that this airfoil in Region I is either fully separated or has a laminar separation bubble.
Surface-oil flow visualization is utilized as a secondary measurement technique for this study in an attempt to validate the aforementioned hypothesis through the identification of separation, as well as boundary layer characteristics. Results of $\alpha = -1^\circ$ are provided in Figure 27. Separation is indicated by the accumulation of oil at $x/c \approx 0.45$ which corresponds well with the pressure distribution location of separation. Downstream of the oil accumulation, the original coat remains untouched since no surface shear stress is present. Near the leading edge the surface is solid white, indicating laminar flow with high velocity. From the surface-oil flow visualization, no remarks about a long laminar separation bubble can be made so another measurement technique needs to be called upon.
To investigate the global flowfield at $\alpha = -1^\circ$ particle image velocimetry is employed, an image of which is provided in Figure 28. The blackened region is the airfoil contour and the grey area is where no data was obtained from an absence of laser light. The particle image velocimetry figure includes contour maps of velocity magnitude, velocity vectors at equally spaced chordwise distances and streamlines to help visualize the flow.

It is clearly evident that this airfoil separates at $x/c \approx 0.5$ as a free shear layer and not a separation bubble. The evidence of the separation bubble in the surface pressure distribution is then due to the extrapolation of the trailing edge point. The pressure at the trailing edge is required to calculate lift, so the last tap on the upper and lower surface are averaged and used as the trailing edge pressure. The magnitude of the trailing edge point has little effect on the lift calculation, but it affects the qualitative analysis of the upper surface pressure distribution.

The separated flow also contains a mild region of recirculation that slightly decreases surface static pressure which gives explanation to the favorable pressure gradient aft of the airfoil.
the separation point apparent in Figure 26. Because particle image velocimetry cannot be obtained near-wall due to image distortion and glare, it is likely that separation occurs slightly upstream of $x/c \approx 0.5$ in agreement with the surface pressure distribution and surface-oil flow visualization.

![Particle image velocimetry image of baseline flow at $\alpha = -1^\circ$ at $Re = 64,000$](image)

**Figure 28.** Particle image velocimetry image of baseline flow at $\alpha = -1^\circ$ at $Re = 64,000$

3.2.1.2 Region II

The second region of distinct flow is from $0^\circ < \alpha < 13^\circ$ (Region II), where the flow experiences large regions of laminar separated flow that severely reduces the performance of the airfoil. Large regions of separated flow are detrimental as they considerably thicken the effective airfoil shape and wake region, and increase the pressure on the upper surface of the airfoil which decreases lift significantly. The location of separation moves forward as the angle of attack is increased because the streamwise adverse pressure gradient is greater. This enlarges the separated region as angle of attack increases which further decreases lift. The loss is most substantial at an
angle of attack of around 10° for this airfoil, thus 10° is selected as the representative angle for Region II since it has large laminar stall and can potentially result in major performance increases.

As in Region I, the plan is to use information about the pressure distribution, surface-oil flow visualization and particle image velocimetry to diagnose the flowfield and characteristics. The surface pressure distribution from Figure 29 shows that the experimental separation point is located at \( x/c \approx 0.2 \) with a considerable portion of the airfoil under separated flow. This point is in approximate agreement with the viscous XFLR5 solution, however, XFLR5 shows slightly higher values, and transition and subsequent reattachment of the separated boundary layer around \( x/c \approx 0.5 \) where the experimental data does not, which explains the higher lift in Figure 23 at \( \alpha = 10^\circ \).

![Figure 29. Baseline pressure distribution of upper surface at \( \alpha = 10^\circ \) at \( Re = 64,000 \)](image-url)
Figure 30 shows the surface-oil flow visualization at $\alpha = 10^\circ$, which confirms that the separation location ($x/c \approx 0.2$) has moved forward from the previous angle. The solid white surface upstream of the separation point supports the laminar flow hypothesis. It is important to note a second line of oil accumulation, at $x/c \approx 0.65$, downstream of the separation point which may be a result of reverse flow accumulation of the circulation region balanced by gravity and surface tension, or evidence of a separation bubble.

The particle image velocimetry image provided in Figure 31 contains the desired answer regarding the secondary accumulation line. As a note, the rectangular block of grey in the upper right hand of the image is a region where no data was taken. The boundary layer is fully separated from the airfoil surface and contains a large recirculation region with no sign of a separation bubble which is in agreement with the findings of Mack et al.$^{10}$ The secondary accumulation line of Figure 30 is merely due to the strong recirculation shown by the backwards pointing velocity vectors of Figure 31.

![Flow direction](image_url)
From Figure 31, separation appears to occur at $x/c \approx 0.25$ which is aft of the prediction from the other measurements, but is likely due to the lack of resolving velocity vectors near the wall.

![Figure 31. Particle image velocimetry image of baseline flow at $\alpha = 10^\circ$ at $Re = 64,000$](image)

### 3.2.1.3 Region III

For $13^\circ < \alpha < 19^\circ$ (Region III), a sudden and dramatic increase in lift is experienced which is almost certainly the result of the formation of a laminar separation bubble near the leading edge based on the results of previous work. Recall from Section 1.3.3.2, these studies assert that over this bubble the boundary layer rapidly transitions to turbulent due to aggravation of boundary layer stability within the free shear layer and reattaches a short distance downstream of the separation point. This new turbulent boundary layer will still separate, but much further downstream due to its resistivity to the adverse pressure gradient. The work of Mack et al. proposes the bubble produces
spanwise “rollers” which provide necessary wall-normal momentum exchange to attach the flow, but also inherently induces unsteadiness.\textsuperscript{10}

The angle of attack of 16° is used to characterize Region III because it represents a value less than the maximum lift coefficient, where there may still be potential for increased lift with flow control. To formally investigate the cause of the reaction of increased lift, Figure 32 is called upon to review the pressure distribution. It is apparent that a short laminar separation bubble does exist which is revealed by the boundary layer separation at $x/c \approx 0.05$ and the subsequent transition and reattachment around $x/c \approx 0.1$ in both the experimental data and the XFLR5 viscous analysis. The airfoil then turbulently separates at $x/c \approx 0.4$.

![Baseline pressure distribution of upper surface at $\alpha = 16^\circ$ at $Re = 64,000$](image)

**Figure 32.** Baseline pressure distribution of upper surface at $\alpha = 16^\circ$ at $Re = 64,000$
Confirmation of these findings is provided in Figure 33 which shows the oil flow visualization of the baseline flow at \( \alpha = 16^\circ \). Laminar separation is evident near the leading edge by the distinct accumulation of oil. Reattachment, indicated by an absence of fluid, is apparent at \( x/c \approx 0.1 \). For reattachment to occur, transition must transpire slightly upstream, as indicated. Lastly, turbulent separation, where a less distinct and uniform accumulation exists, can be seen at \( x/c \approx 0.4 \). While the location of these phenomena is placed qualitatively, there is strong agreement between the pressure distribution and oil visualization of the overall character.

![Figure 33. Surface-oil flow visualization of the baseline flow at \( \alpha = 16^\circ \) at \( Re = 64,000 \)](image)

Figure 34 provides particle image velocimetry of the flowfield situation at \( \alpha = 16^\circ \). It can be seen that turbulent separation occurs at \( x/c \approx 0.45 \); however, the laminar separation bubble cannot be seen due to particle image velocimetry near-wall resolution. This demonstrates the small geometry of the short bubble.
Figure 34. Particle image velocimetry image of baseline flow at $\alpha = 16^\circ$ at $Re = 64,000$

From this study, it is relatively simple to deduce the cause of the reducing lift within Region III, where a lift coefficient plateau is not achieved. Initially, when the bubble forms at the lower angles of Region III it is small because transition occurs just beyond separation and reattaches quickly. As the angle is increased, separation occurs further upstream which creates a longer laminar separation bubble that is weaker and disrupts the lift.

3.2.1.4 Region IV

Within Region IV ($\alpha > 19^\circ$) the adverse pressure gradient becomes so extreme that laminar separation moves further forward to the leading edge where transition to turbulence does not occur rapidly enough and boundary layer turbulence does not entrain enough freestream momentum to reattach the flow. The result is a significant decrease in lift. For Region IV, $\alpha = 20^\circ$ is used as the studied angle because it is the first angle within this region, and is thus where flow control has the most potential to extend the
range of angle of attack. Per usual, the pressure distribution is studied first in Figure 35. It is easy to see that the airfoil is completely separated which is made obvious by the flat line pressure distribution, and follows closely the XFLR5 viscous prediction. The separation point occurs very close to the leading edge, the exact location of which is difficult to define but not necessary for the continuation of the study.

![Baseline pressure distribution of upper surface at α = 20° at Re = 64,000](image)

**Figure 35. Baseline pressure distribution of upper surface at α = 20° at Re = 64,000**

The results of the oil flow test at α = 20° are provided in Figure 36 which shows the previously mentioned separation with a clear accumulation of oil at the leading edge and untouched oil at all locations downstream. All non-uniformity present in the image is the result of gravitational forces and recirculation shear.
Figure 36. Surface-oil flow visualization of the baseline flow at $\alpha = 20^\circ$ at $Re = 64,000$

An image of particle image velocimetry of the flowfield is shown in Figure 37. The figure provides little new information other than to confirm the previous conclusion of massive separation of the boundary layer near the leading edge of the airfoil. The lack of a suction peak along with the high static pressure on the upper surface of the airfoil from separation contributes to significant loss of lift in this region.
3.2.1.5 Summary

The major results of this section are to define and confirm the hypothesis of four distinct regions of airfoil aerodynamics. Particle image velocimetry, surface pressure distributions and surface-oil flow visualization are employed to investigate the nature of the airfoil characteristics in each region by defining the location of separation and type of boundary layer present, as well as any other interesting flow phenomena present.

The four regions identified are weak laminar separation (Region I, $\alpha < 0^\circ$), moderate laminar separation (Region II, $0^\circ < \alpha < 13^\circ$), laminar separation bubble (Region III, $13^\circ < \alpha < 19^\circ$), and strong leading edge laminar separation (Region IV, $\alpha > 19^\circ$). The definition of the endpoint angles of these regions are not set in stone, they merely divide the four regions. Looking back at Figure 23 and Figure 24, it is very evident that a smooth curve can be drawn through Regions I, II and IV which shows how the decrease
in lift and increase in drag is simply the result of the laminar separation point moving forward. In Region III, the transition of the boundary layer within the separated shear layer reattaches the flow such that turbulent separation is the limiting feature of the airfoil. Thus it can be concluded that Region III represents a portion of a turbulent lift and drag curve.

The purpose of this section has primarily been used to provide background of the type of flow present and motivate the need for flow control to improve the aerodynamics within each distinct region. Through investigation, it is apparent that all separations occur with a laminar boundary layer, thus a tripping device would prove to be very useful. So as not to defeat the purpose of a laminar airfoil, the transition to turbulence should occur just before the separation point.

3.2.2 Flow Control

Since the airfoil exhibits distinct behavior in each of these four regions it is therefore possible that the most effective flow control method may also be distinct within each region. Consequently, at each representative angle of attack a range of blowing ratios and suction strengths are investigated in order to optimize the airfoil lift. Additionally, zigzag tape is utilized in order to understand the effectiveness of the tape as a passive flow control technique.

Blowing ratio is used as the measure of actuation level (instead of momentum coefficient) because the greatest concern here is the flow physics induced which is a function of blowing ratio. The local blowing ratio is used because it represents the normal-to-crossflow velocities that affect the jet physics which are briefly described in
Section 1.3.4.3, rather than the freestream blowing ratio which is less relevant. The blowing ratio \((BR)\) is defined in Equation 12 as the ratio of the normal jet velocity to the local freestream velocity.

\[
BR = \frac{U_{\text{holes}}}{U_{\text{f}}}
\]  

Equation 12

The local freestream velocity is obtained from the surface pressure distribution and the normal jet velocity is calculated by assuming conservation of mass through the rotameter and out of a fixed number of equally sized fluidic actuator holes. The suction strength is measured in terms of the volume flow rate traveling through the rotameter from the same number of equally sized fluidic actuator holes, and consequently the “effectiveness” of each method can be compared according to the correlation of volume flow rate required and lift change. Recall, the dimensions for the zigzag tape are taken from Mack \textit{et al.}\textsuperscript{10} and scaled with airfoil chord (see Section 2.5), and the location of all control occurred at 5% chord.

A brief investigation of blowing is performed to demonstrate the independence of each blowing hole and to confirm the above findings. Figure 38 provides a qualitative infrared image of 20\% of the span and 20\% of the chord, showing the coherence and independence of the structures as they travel downstream of the jet in cross flow at a hole spacing of 10.67\(d\). The chordwise location of the jets has been marked by a dashed line and the hole locations are represented by the equally spaced dots. The hot streaks originate from the flow control holes and continue downstream despite the fact that the
jet fluid is lower temperature. The hotter streaks appear because the spanwise vortices which form from a jet in crossflow are effective at entraining the hotter freestream flow into the boundary layer and convecting it downstream. Distinct structures are present until $x/c \approx 0.25$, wherein diffusion through viscosity and mixing dampen the effect.

![Flow direction diagram](image)

**Figure 38. Infrared image of 20% span and 20% chord, showing the coherent and independent structures downstream of the normal jets at $\alpha = 10^\circ$ and $BR = 0.29$**

A summary of the flow control methods is presented in the following. Table 1 summarizes the optimized blowing parameters, lift increase, drag decrease and figure of merit for the four representative angles. Recall, that a figure of merit greater than one means that energy of actuation is worth effort and should be employed. The range of local blowing ratios scanned is from 0 to 3. In some instances a higher blowing ratio does provide a minimal addition of lift, yet a lower blowing ratio is taken to be optimal as
the marginal increase in lift does not justify the significant increase in required volume flow rate.

Table 1. Summary of the optimized blowing parameters and performance measures at $Re = 64,000$

<table>
<thead>
<tr>
<th>Region</th>
<th>Angle</th>
<th>Blowing Ratio</th>
<th>Volume Flow (SCFH)</th>
<th>% Lift Increase</th>
<th>% Drag Decrease</th>
<th>Figure of Merit</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>-1°</td>
<td>1.63</td>
<td>19</td>
<td>177.2</td>
<td>-0.1</td>
<td>0.40</td>
</tr>
<tr>
<td>II</td>
<td>10°</td>
<td>0.29</td>
<td>6</td>
<td>114.1</td>
<td>77.2</td>
<td>8.62</td>
</tr>
<tr>
<td>III</td>
<td>16°</td>
<td>2.02</td>
<td>60</td>
<td>-0.8</td>
<td>-72.5</td>
<td>0.07</td>
</tr>
<tr>
<td>IV</td>
<td>20°</td>
<td>2.87</td>
<td>60</td>
<td>43.2</td>
<td>32.1</td>
<td>0.52</td>
</tr>
</tbody>
</table>

It is apparent from Table 1 that blowing improves lift significantly in Regions I and II, and moderately in Region IV. Blowing also significantly improves drag in Region II and moderately in Region IV. The volume flow required for Region II is also significantly less than in all other regions. The three of these factors combine to provide the figure of merit which states that in Region I, III and IV, the actuation is not worth the effort from an energy standpoint because it is less than one. On the other hand, blowing should definitely be utilized in Region II where the figure of merit is well above one. This indicates that major airfoil performance is gained with little input. Region III does not benefit from the use of blowing where the performance actually decreases with the use of any amount of blowing. During the laminar leading edge separation of Region IV, the lift and drag are reasonably improved but the required volume flow is so high that the
figure of merit is below one and the control is not energy effective. Overall, Regions I, II and IV see improvement in airfoil performance, but only Region II displays enough improvement from an energy standpoint to justify control. The results hint that in Region II, some instability is apparent and only needs a small disturbance to instigate change; however, all other regions require significant actuation to obtain change.

Suction, a summary of which is given in Table 2, is only consistently effective in Region IV. The table does not provide the values of performance for the other regions because there is either no improvement, or the improvement is not consistent enough to consider it to be a reliable flow control method. The fact that there is no improvement is justifiable since suction removes the boundary layer, making it more laminar, which does nothing to prevent laminar separation.

Table 2. Summary of the optimized suction parameters and performance measures at $Re = 64,000$

<table>
<thead>
<tr>
<th>Region</th>
<th>Angle</th>
<th>Volume Flow (SCFH)</th>
<th>% Lift Increase</th>
<th>% Drag Decrease</th>
<th>Figure of Merit</th>
</tr>
</thead>
<tbody>
<tr>
<td>IV</td>
<td>20°</td>
<td>30</td>
<td>44.0</td>
<td>36.3</td>
<td>0.86</td>
</tr>
</tbody>
</table>

Again, as with blowing, for suction there is moderate lift increase and drag decrease but due to the cost of the high power required to actuate the device, the figure of merit recommends not doing suction to conserve energy. Suction, however, is more beneficial than blowing so it makes an excellent flow control device to extend the range of angle of attack for an aircraft to perform some high performance maneuvers or use in a time of
need. It is similar to an afterburner in that an extreme amount of inefficient power is used to enhance performance only when necessary. The same can be said about blowing and suction in Region IV.

Passively controlling the flow with zigzag tape provides some interesting results, as seen in Table 3. The performance enhancement of zigzag take is beneficial in all regions except in Region III. The reduction of drag in Region I is within measurement error. The figure of merit in this case is just a ratio of lift-to-drag since there is no actuation power required. Region I and IV see small improvements from the zigzag tape. There is a significant increase in Region II of the baseline performance and over the blowing performance because the lift is slightly higher and at a cost of no actuation. However, zigzag tape is detrimental in Region III as it apparently disrupts the laminar separation bubble. Zigzag tape, unlike blowing or suction, cannot be removed at certain angles which is a poor quality of the flow control method.

Table 3. Summary of the zigzag performance measures at $Re = 64,000$

<table>
<thead>
<tr>
<th>Region</th>
<th>Angle</th>
<th>% Lift Increase</th>
<th>% Drag Decrease</th>
<th>Figure of Merit</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>-1°</td>
<td>6.4</td>
<td>-0.3</td>
<td>1.07</td>
</tr>
<tr>
<td>II</td>
<td>10°</td>
<td>121.6</td>
<td>77.3</td>
<td>9.77</td>
</tr>
<tr>
<td>III</td>
<td>16°</td>
<td>-1.1</td>
<td>-18.7</td>
<td>0.83</td>
</tr>
<tr>
<td>IV</td>
<td>20°</td>
<td>6.8</td>
<td>8.2</td>
<td>1.16</td>
</tr>
</tbody>
</table>
Figure 39 compares $C_L$ versus $\alpha$ of the flow control and baseline results at the representative angles of attack for each region. The lift-drag curve is shown in Figure 40 which includes colored lines to show where the flow control angles emanate from. The results of the control studies (zigzag tape at 2% chord and plasma actuator at 40% chord) by Mack et al.\textsuperscript{10} are also included to help show the trends of flow control and validate results. An important aside to make here is that the baseline measure by Mack et al. for control studies was with a slightly different airfoil, so the laminar separation bubble lift coefficient plateau is not as significant and more on the order of what is measured for the baseline of this study. It is apparent that both control methods by Mack et al. show the same trend as blowing in Regions I-III, but show no improvement in Region IV which could be the result of actuation location. Mack et al. also shows a significant drop in lift in Region III for zigzag tape and a minor drop for plasma actuation, where the laminar separation bubble is suggested to be disturbed.

Both Figure 39 and Figure 40 express the overall effect of each control method for this case. Suction is not a desirable option because it is inconsistent and only benefits Region IV. Zigzag tape is also not the best method because although it is better than blowing in Region II, it is harmful in Region III and cannot be turned off, and has little effect in Region I and IV. Blowing is clearly the best option because it can be adjusted and has a beneficial effect in three of the four regions. Blowing has the potential to create a smooth lift curve similar to a turbulent boundary layer lift curve which is demonstrated by the control of Mack et al. Blowing can also extend the range of angle of attack at $Re = 64,000$. Further discussion regarding the flow physics responsible for these
results, through the use of surface pressure measurements, particle image velocimetry and boundary layer profiles, are subsequently provided for each of the four representative angles.

Figure 39. $C_L$ vs. $\alpha$, $Re = 64,000$, control comparison of NACA 64-3-618

Figure 40. $C_L$ vs. $C_D$, $Re = 64,000$, control comparison of NACA 64-3-618
3.2.2.1 Region I (\(\alpha = -1^\circ\), Weak Laminar Separation)

Here, control of Region I will be studied with the same measurement techniques from the baseline case of Section 3.2.1.1 and will compare the results between control and baseline. A comparison of surface pressure measurements, derived by static pressure taps along the length of the airfoil, is provided in Figure 41. The normalized local velocity, derived from a hot-film at \(y/c \approx 0.02\) from the upper surface of the airfoil, is provided in Figure 42. Hot-film anemometry is introduced because of its ability to capture flow statistics near the surface where information about the flow is most important and to corroborate the surface pressure data. Figure 43 shows the freestream normalized mean velocity for the blowing configuration derived from the particle image velocimetry investigation. Note that the bunching and subsequent gap of the streamlines in Figure 43 is non-physical and is the result of the meshing of the two images. A figure of the particle image velocimetry for zigzag tape is not produced because minimal change in performance is noted. Suction has been left out completely due to its lack of effectiveness. Unlike before, these three techniques will be analyzed simultaneously because it is easier to diagnose the airfoil physics from many angles at once.
Figure 41. Baseline vs. control pressure distribution of upper surface at $\alpha = -1^\circ$ at $Re = 64,000$, $BR=1.63$

Figure 42. Baseline vs. control normalized velocity at $y/c \approx 0.02$ from upper surface at $\alpha = -1^\circ$ at $Re = 64,000$, $BR=1.63$
Figure 43. Particle image velocimetry image of blowing flowfield at $\alpha = -1^\circ$ at $Re = 64,000$, $BR=1.63$

Static pressure measurements, hot film anemometry and the particle image velocimetry corroborate that blowing with $BR = 1.63$ increases the local velocity along the length of the chord and slightly delays separation to $x/c \approx 0.55$. Both the surface pressure measurements and particle image velocimetry indicate the reattachment of flow at the trailing edge of the airfoil. This indicates the presence of a separation bubble, which is the reason for an increased lift of 177.2%. The result of a 0.1% increase in drag is within measurement error and likely not correct. The formation of the laminar separation bubble is the result of some physics that only the blowing induces, and not the zigzag tape. It is apparent that blowing physics reduce the separation height, which increases circulation and upper surface velocity. The velocity increase is seen in the green circles of Figure 41 and Figure 42, and the larger suction peak and overall velocity in Figure 43.
It is surprising that the zigzag tape has no effect on the flow since it was anticipated to induce transition of the laminar baseline flow thereby suppressing separation and positively impacting lift. The major cause that is suspect for this result is that the boundary layer is disturbed by zigzag tape, but because the baseline flow is very viscous and has a significant distance of a favorable pressure gradient to travel, the effects are damped out completely before the adverse pressure gradient can amplify the disturbances into anything effective.

The moderate blowing ratios required to induce change at this angle appear to create large enough vortices and disturbances to withstand the favorable pressure gradient and affect the flow. The moderately-sized vortices entrain freestream flow into the boundary layer to increase velocity and delay separation. The larger disturbances last through the suppressing favorable pressure gradient to transition the flow near separation resulting in a thinner separated region and an increase in lift. The transition process takes a relatively long distance which explains why the separation bubble is longer. To test these ideas and further investigate the boundary layer characteristics and flow control mechanisms for blowing and zigzag tape, the hot-film is used to probe the boundary layer.

A boundary layer profile at $x/c \approx 0.15$, which is in a region of the favorable pressure gradient and 10% downstream of the flow control, can be seen in Figure 44. For this figure, the velocity profile is provided on the left which has been normalized with the boundary layer edge velocity and plotted against $y/c$. The profile on the right is the local turbulence intensity, $Tu$, and again plotted against $y/c$. The boundary layer is investigated for the baseline, blowing and zigzag tape configurations. Recall from Figure 38, the
normal jet in crossflow has coherent structures for some distance downstream. Accordingly, the boundary layer is investigated at two spanwise locations, in order to understand the spanwise uniformity of the blowing configuration. Defining $z$ to be the spanwise distance from an actuation hole, the first location is downstream of the blowing hole at the same spanwise location ($z/s = 0$, on hole) and the second location is between two adjacent holes ($z/s = 0.5$, off hole). A similar profile at $x/c \approx 0.4$ is presented in Figure 45, which is 35% downstream of the flow control and just beyond the favorable pressure gradient. This will demonstrate the effect of pressure gradients on the boundary layer and control mechanisms.

![Figure 44. Baseline vs. control boundary layer velocity and turbulence profiles at $x/c \approx 0.15$ at $\alpha = -1^\circ$ at $Re = 64,000$, $BR=1.63$](image)

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Figure 45. Baseline vs. control boundary layer velocity and turbulence profiles at $x/c \approx 0.4$ at $\alpha = -1^\circ$ at $Re = 64,000, BR=1.63$

From the aforementioned figures, only a little information can be dissected about the character of the boundary layer. It is evident that blowing at $x/c \approx 0.15$ increases the on and off hole velocities equally from that of the baseline, creating a fuller and more stable boundary layer that is less susceptible to separation. The equality between on and off hole velocities indicates some degree of spanwise uniformity at this location. It is interesting to note that the zigzag tape boundary layer also shows a somewhat fuller velocity profile compared to the baseline flow, which may be due to the beginning of entrainment of the freestream due to some minor flow disturbances. Another point of interest is that the turbulence intensity levels are constant and similar between the control and baseline flows meaning that no turbulence has been generated within the boundary
layer at this point due to control. The low levels of turbulence intensity indicate laminar and steady flow and neither control case has perturbed turbulence.

The information from $x/c \approx 0.4$ tells more of an interesting story. The zigzag tape values have fallen on top of the baseline values supporting the theory that the zigzag tape has some effect near the leading edge, but is then eliminated downstream via the favorable pressure gradient. It is also noticeable that the turbulence intensity is beginning to grow near the surface indicating the onset of some unsteadiness. The velocity profile of the on hole compared to the off hole blowing case is fuller which shows that the entrainment due to rollers dissipates downstream off of the hole, but remains strong on hole.

Before stating the final conclusions of the flow control in Region I, the power spectral density can be checked to see what the strongest frequency content is, imply the size of the structures generated by the control, and see if turbulence or transition is present. The hot-film is employed to obtain the power spectral density which is performed at $y/c \approx 0.002$, a location well within the boundary layer. The results are found in Figure 46 and Figure 47 for $x/c \approx 0.15$ and 0.4, respectively.

The power spectrum at 15% chord confirms that the flow is laminar due to the low energy at high frequency. The off hole blowing has higher energy content in the higher frequencies indicating that blowing has induced some small structures between the holes. In the baseline flow, there are two peaks of higher energy content at 25 Hz and 65 Hz which correspond to reduced frequencies of approximately 0.56 and 1.48. Reduced frequency $(f_c/U_\infty)$ is a way to determine the number of structures present on the airfoil at
any given instant. Thus, there two separate sets of structures that occur, one that is larger in scale than the airfoil itself, and the other occurs relatively often on the airfoil may be associated with smaller scale structures on the airfoil surface. Based on the findings of Mack et al.\textsuperscript{10} and Brehm et al.\textsuperscript{12}, it is suggested that the higher frequencies correspond to wall-bounded shear layer instabilities that coincide with the “rollers” of spanwise vortices. A theory on the peak of low frequency energy may be related to wake shedding.

This could be beneficial information for those who wish to exploit these structures with unsteady flow control. At $x/c \approx 0.15$, blowing appears to have no effect upon either of these structure sets; however, zigzag tape shifts both structures to higher frequencies. It is suggested that the small disturbances of the zigzag tape disrupt the present structures on the airfoil.

At $x/c \approx 0.4$, both sets of structures reappear at their original frequency for the zigzag case due to the suppression of the effect of the zigzag tape from the favorable pressure gradient and the re-emergence of the baseline structures. The on hole blowing location appears to be displaying the onset of transition from the high energy at high frequencies. The off hole location also sees a slight increase in higher frequency content. This is indicating that transition starts on hole and starts expanding outward downstream.
Figure 46. Power spectral density of the boundary layer at $x/c \approx 0.15$ and $y/c \approx 0.002$ at $\alpha = -1^\circ$ at $Re = 64,000$, $BR=1.63$

Figure 47. Power spectral density of the boundary layer at $x/c \approx 0.4$ and $y/c \approx 0.002$ at $\alpha = -1^\circ$ at $Re = 64,000$, $BR=1.63$
Based on these general observations, the mechanisms by which the blowing is effective and zigzag tape is ineffective are proposed as follows. While normal blowing adds no direct streamwise momentum, the boundary layer is energized through moderate three-dimensional structures which entrain higher momentum freestream flow. These structures are formed by the moderate blowing ratio which creates vortices behind it and moves freestream momentum into the boundary layer creating a fuller velocity profile. The more full velocity profile slightly delays separation and increases upper surface velocity. The moderate disturbances induced by blowing last through the favorable pressure gradient and become amplified in the adverse pressure gradient which transition to turbulence at separation to form a laminar separation bubble near the trailing edge. This increases the circulation of the airfoil which results in a higher lift coefficient.

For the zigzag tape, the results of the boundary layer investigation are clear that, at least 0.1c downstream of the control, transition does not occur as a result of suppression of the small disturbances from the favorable pressure gradient. The location of the flow control is a likely cause of this phenomenon. Had the zigzag tape been located just upstream of the baseline separation location (about 50% chord) in a region of adverse pressure gradient, it is proposed that transition would have occurred and separation reduced.

3.2.2.2 Region II ($\alpha = 10^\circ$, Moderate Laminar Separation)

As discussed in Section 3.2.1.2, baseline separation for $10^\circ$ is at $x/c \approx 0.2$, which occurs via a laminar boundary layer. The objective is to study the physical mechanisms
that improve lift by 114.1% for blowing and 121.6% for zigzag tape, and reduce drag by 77.2% and 77.3% for each, respectively.

Static pressure measurements (Figure 48), hot-film anemometry (Figure 49) and particle image velocimetry (Figure 50 and Figure 51) agree that blowing with $BR = 0.29$ and using zigzag tape increases the local velocity along the length of the chord, delays separation and dramatically increases the lift. It also appears as though the flow physics inducing the change are the same for both blowing and zigzag tape, because all three measurement methods show nearly the same character for both controls. It is difficult to discern the location of separation from the surface pressure measurement and particle image velocimetry. From the hot-film slice, it is convincing that the separation location is roughly located at $x/c \approx 0.45$ with transition occurring at $x/c \approx 0.6$ and reattachment at a location further downstream. Recall, the actual separation location must occur slightly upstream of what the hot-film predicts.

The major take away from the comparison is not the location of separation, rather, that both blowing and zigzag tape produce the same results. From the surface pressure distribution, both also separate, transition and reattach in similar locations, meaning the same physics is present and initiating the increase in lift. The surface pressure distribution also follows a similar trend as the XFLR5 viscous solution of Figure 29. Because the viscous solution diverges significantly from the inviscid solution it can be concluded that this is a long laminar separation bubble, using the guidelines set out by Hägmark$^{14}$ who suggested a divergence indicates such. The separated region, although it is rather long, is fairly thin and difficult to pull from Figure 50 and Figure 51.
Figure 48. Baseline vs. control pressure distribution of upper surface at $\alpha = 10^\circ$ at $Re = 64,000$, $BR=0.29$

Figure 49. Baseline vs. control normalized velocity at $y/c \approx 0.02$ from upper surface at $\alpha = 10^\circ$ at $Re = 64,000$, $BR=0.29$
The hot-film is made use of to probe the boundary layer at 12%, which is approximately the suction peak, shown in Figure 52. At $\alpha = 10^\circ$ at $x/c \approx 0.12$, the
velocity profiles are all very similar when compared to the baseline. The turbulence intensity is generally higher at $\alpha = 10^\circ$ compared to what is observed at $\alpha = -1^\circ$ and there is some indication of disturbances in the flow for baseline and blowing because of the higher values near the surface. Oddly, the zigzag tape shows a suppression of disturbances within the flow at 12% chord. These results are unexpected, as it was anticipated that both the blowing and zigzag control would cause the flow to transition immediately, however, it must occur further downstream where an adverse pressure gradient is present. The cause of suppression of disturbances for zigzag tape is suggested to be a result of the small scale vortices of the zigzag tape disrupting the baseline flow structures present.

In order to understand how flow control affects the boundary layer downstream, a similar investigation of the boundary layer is performed at $x/c \approx 0.30$, well downstream of the suction peak in a moderate adverse pressure gradient before the control separated region. These results are presented in Figure 53. The baseline flow is obviously separated at 30% chord. The controlled flow looks laminar due to the low turbulence intensity values for the control cases, so a trip still has not occurred. The blowing cases have a fuller velocity profile from spanwise vortices induced by a low blowing ratio. The blowing turbulence intensities have decreased significantly and the zigzag turbulence has increased slightly near the surface, which is also interesting as it is expected they trend in the same direction. It appears as though the zigzag tape Removes some baseline structures in place of its own disturbances, which then grow downstream. This was also seen in Region I. Oppositely, the blowing creates its own structures which are
suppressed downstream, possibly from diffusion of the vortices in the adverse pressure gradient.

Figure 52. Baseline vs. control boundary layer velocity and turbulence profiles at $x/c \approx 0.12$ at $\alpha = 10^\circ$ at $Re = 64,000$, $BR=0.29$

Figure 53. Baseline vs. control boundary layer velocity and turbulence profiles at $x/c \approx 0.3$ at $\alpha = 10^\circ$ at $Re = 64,000$, $BR=0.29$
The power spectral density is taken for the three control configurations, namely blowing at \( z/s = 0 \) and 0.5 and zigzag tape, as seen in Figure 54 for 12\% chord and Figure 55 for 30\% chord. The baseline, since it is still attached at 12\% chord, has been included in Figure 54. Similarities between these results and those shown for \( \alpha = -1^\circ \) are prevalent; large scale, low frequency (25 Hz) structures are present. The high frequency structures around 65 Hz that are apparent at \(-1^\circ\) for all cases has been attenuated near the leading edge, but maintain their presence downstream. The zigzag tape again shows the suppression of all large scale structures at 12\% chord but then again re-emerges at 30\% chord, similar to Region I and the boundary layer profile above. Zigzag tape at 30\% chord also shows distinct high frequencies of high energy, which may be transitional flow.
Figure 54. Power spectral density of the boundary layer at $x/c \approx 0.12$ and $y/c \approx 0.002$ at $\alpha = 10^\circ$ at $Re = 64,000$, $BR=0.29$

Figure 55. Power spectral density of the boundary layer at $x/c \approx 0.3$ and $y/c \approx 0.002$ at $\alpha = 10^\circ$ at $Re = 64,000$, $BR=0.29$
Based on these general observations, the mechanisms by which blowing and zigzag tape are effective are proposed as follows. The mechanism for normal blowing and zigzag tape at $\alpha = 10^\circ$ are slightly different than at $\alpha = -1^\circ$. In recollection, in Region I, normal blowing energizes the boundary layer through moderate three-dimensional structures which entrain higher momentum freestream flow. Because the blowing ratio at $\alpha = 10^\circ$ is much lower, the three-dimensionality effects are much smaller.

As mentioned before, zigzag tape produces small three-dimensional vortex pairs which are small, yet frequent. Within a favorable pressure gradient, the size of these vortices is reduced, and as seen in Region I eliminated completely via an extended region of favorable pressure gradient. If the gradient is short or shallow enough, as in this region, the vortices outlast the favorable pressure gradient and begin to grow in the adverse pressure gradient. As they grow, they can then reduce separation from slight momentum exchange with the freestream and also transition the boundary layer at separation to create a laminar separation bubble. Because these are small and frequent, they disrupt other structures present which explains the damping of large scale structures in Figure 54. The significant level of small scale structures in Figure 55, which are likely the small vortices that have grown in the adverse pressure gradient.

It is known that blowing at low ratios creates a small shell, similar to the one in Figure 9, and it is proposed that the discrete shell also induces small three-dimensional vortices (streamwise and spanwise) similar to the zigzag tape. This is proposed because the results of both blowing and zigzag tape are so similar. The shells are less frequent because of the distance between each blowing hole, so they impact the large scale
structures much less, shown in Figure 52 and Figure 54. It is also seen in the power spectrum that there is slightly more high frequency content at the on hole location supporting the idea of generation of small vortices along the hole but not between due to lack of three-dimensionality. It is interesting to note that a much smaller blowing ratio is employed in this configuration, yet the effect is much more dramatic because the decreased region of favorable pressure gradient allows the smaller leading edge vortices to propagate further downstream. The small vortices do not last in the Region I pressure gradient, which is why larger structures need to be induced to travel a long distance downstream without being suppressed.

For both control applications, the disturbances from the vortices become amplified and create a transitional boundary layer which then separates, creates turbulence, and forms a long laminar separation bubble. This bubble formation reduces the thickness of the separated region which increases circulation in both the blowing and zigzag tape cases. This is intimately connected to the significant velocity increase over the upper surface. This is sensible since not much velocity is added by blowing, zigzag tape or the small entrainment associated with their structures. Because the vortices of the zigzag tape are more frequent and uniform than those from the blowing shell, slightly more freestream velocity is entrained and transition is induced faster, thus the lift is higher.

Hysteresis effects are evident while utilizing normal blowing. After initializing control and validating that a positive effect had been achieved, the blowing could be turned off completely and the lift increase persisted for an extended period. This process was repeated numerous times; the shortest recorded time was 5 minutes, and the longest

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interval was ended prematurely at the 15 minute mark. This result is significant; as the long laminar separation bubble may theoretically be maintained indefinitely with very minimal control by simply actuating for a few seconds every couple of minutes. For this reason alone, normal blowing is the more attractive control configuration, as this proves to be extremely valuable in closed-loop control applications.

3.2.2.3 Region III ($\alpha = 16^\circ$, Laminar Separation Bubble)

This region is suggested to have a laminar separation bubble which maintains a natural mechanism of flow control and increases performance significantly without the use of external work. Measured values of surface pressures are provided in Figure 58 to show the endeavor in improving an already well-controlled flow. As evidenced in this $C_P$ plot, the $C_L$ plot in Figure 39 and the $C_D$ plot in Figure 40, all attempts at improving the lift and drag through passive and active flow control techniques are fruitless. The attempts even reduced lift and increased drag no matter what amplitude of control is implemented. As noted by Mack et al.\textsuperscript{10}, because of its very beneficial effect on the airfoil performance the occurrence of a laminar separation bubble, especially when it resides near the leading edge, is a natural flow control mechanism. From this study, it also demonstrates the resilience of the separation bubble in that blowing or zigzag tape does not prevent the formation of the bubble.

It is an interesting point to note that the baseline surface pressure distribution falls directly between blowing and zigzag tape, but the performance it still better with the baseline distribution. A hypothesis on the cause is that the natural size and location of the bubble induces the optimal structures and provides the best performance. As zigzag
tape is used it reduces peak velocity and extends the length of the laminar separation bubble, thus increasing drag and reducing lift. For blowing, the location of the separation bubble changes which also increases drag and decreases lift. It is best to just leave the laminar separation bubble alone, thus having a passive, permanent flow control method is not advisable.

![Figure 56. Baseline vs. control pressure distribution of upper surface at \( \alpha = 16^\circ \) at \( Re = 64,000, BR=2.02 \)](image)

Because the natural control is deemed optimal, particle image velocimetry and hot-film investigations of the controlled flow are not done. Boundary layer measurements of the baseline via hot-film are employed in order to further study the flow associated with the laminar separation bubble. The boundary layer is investigated at \( x/c \approx 0.07 \) and 0.4. According to the surface pressure measurements, the former location should be just
within the laminar separation region and experiencing transitional flow. The latter location should be fully turbulent and just before the turbulent separation region. Figure 57 compares the velocity profile and turbulence intensity at the two locations, while Figure 58 compares the power spectral density.

![Figure 57. Baseline boundary layer velocity and turbulence profiles at x/c ≈ 0.07 and 0.4 at α = 16° at Re = 64,000](image)

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Figure 58. Power spectral density of the boundary layer at $x/c \approx 0.07$ and 0.4 and $y/c \approx 0.002$ at $\alpha = 16^\circ$ at $Re = 64,000$

While the boundary layer investigation at previous angles of attack is somewhat ambiguous, the profiles measured at $\alpha = 16^\circ$ are easily distinguishable. From Figure 57, the location near the laminar separation point ($x/c \approx 0.07$) does appear to be very full and has low turbulence intensity, indicating laminar flow. There is a small region of higher intensity near the surface indicating possible transition. There is no indication of separation which is a curious thought. However, recall that the separation bubble is very thin; therefore an indication of it is difficult unless data is taken even closer to the wall. The author did not have enough courage to bring the hot-film any closer to the wall, so this hypothesis cannot be confirmed, nor denied. The boundary layer profile at $x/c \approx 0.4$ shows a velocity profile on the verge of an inflection point which leads to separation not far downstream. The turbulence intensity is also very high, indicating turbulence.
The power spectrum at $x/c \approx 0.07$ is similar to what is observed at previous angles of attack, namely a distinguishable frequency observed at around 25 Hz, and afterwards a quick decrease in energy at higher frequency bands. The power spectrum at $x/c \approx 0.4$, however, is different than other spectrum observed thus far. While the blowing configuration at $\alpha = -1^\circ$ does have additional energy at higher frequencies, it is appreciably lower than what is observed here. For $\alpha = 16^\circ$, at lower frequencies a higher energy is measured, and this energy remains nearly constant until around 500 Hz, after which the turbulent Kolmogorov -5/3 spectrum\(^{41}\) is observed.

3.2.2.4 Region IV ($\alpha = 20^\circ$, Strong Leading Edge Laminar Separation)

Within Region IV, the airfoil under baseline conditions has laminar separation at the leading edge with a significant loss of lift compared to Region III. The surface pressure measurements comparing the baseline case to blowing, suction and zigzag tape is provided in Figure 59. It is readily apparent that zigzag tape is ineffective, which is sensible since the flow is separating before the zigzag tape. Blowing and suction, however, provide significant lift enhancement as shown by the low surface pressures in the front portion of the airfoil. With a blowing ratio of 2.87 (60 SCFH), the lift is increased by 43.2% and the drag is decreased by 32.1%. Similarly, with suction of 30 SCFH, the lift is increased by 44.0% and the drag is decreased by 36.3%.

The hot-film slice of Figure 60 shows the similar magnitudes and character between blowing and suction, but is a poor designator of locations because the data is taken outside of the boundary layer, so it does not see laminar or turbulent separation. Both particle image velocimetry (Figure 61 and Figure 62) and surface pressure measurements
agree that turbulent separation occurs at $x/c \approx 0.45$ when either blowing or suction is applied, but again there is no presence of laminar separation.

The mechanisms induced by blowing and suction are presumed to be different; however, the surface pressure measurements are identical, and particle image velocimetry and hot-film data indicates nearly the same flowfield, with suction showing slightly more circulation and a less extreme separation angle at half of the required mass flow for blowing. To investigate further details, the typical hot-film boundary layer probe and power spectrum is discussed in the following paragraphs.

Figure 59. Baseline vs. control pressure distribution of upper surface at $\alpha = 20^\circ$ at $Re = 64,000$, $BR=2.87$, Suction = 30 SCFH
Figure 60. Baseline vs. control normalized velocity at $y/c \approx 0.02$ from upper surface at $\alpha = 20^\circ$ at $Re = 64,000$, $BR=2.87$, Suction = 30 SCFH

Figure 61. Particle image velocimetry image of blowing flowfield at $\alpha = 20^\circ$ at $Re = 64,000$, $BR=2.87$
Figure 62. Particle image velocimetry image of suction flowfield at $\alpha = 20^\circ$ at $Re = 64,000$, 30 SCFH

The boundary layer probing is done at both $x/c \approx 0.1$ and $x/c \approx 0.37$. The results of the boundary layer profiles at $x/c \approx 0.1$, which is just aft of the suction peak, are illustrated in Figure 63. The baseline and zigzag are clearly separated, as expected. Suction is showing an inflection point which is the result of the suction-induced adverse pressure gradient that reduces the velocity in the boundary layer. The effect of suction at this location clearly creates downstream turbulence as indicated by the high turbulence intensity. The boundary layer velocity of blowing both on and off hole looks very full. The blowing boundary layers are smaller than their suction counterpart. The blowing on hole location shows turbulent flow by the high turbulence intensity, but the off hole location does not show turbulence.
Figure 64 provides the results of the boundary layer study at the downstream location ($x/c \approx 0.37$), which is just ahead of the separation location. Both the baseline and zigzag tape data have been left out since they provide no meaningful information. The velocity profiles for both suction and blowing are very shallow which signifies the onset of separation. The turbulence intensity values are also high for both indicating fully turbulent flow. The cause for the less aggressive suction departure angle may be the result of suction pulling the stagnation point closer to the leading edge from the lower surface to shift circulation, thus minimizing the separation angle.

The results of the power spectrum only confirm the conclusion of turbulence, which is shown in Figure 65.

![Graph showing baseline vs. control boundary layer velocity and turbulence profiles at $x/c \approx 0.1$ at $a = 20^\circ$ at $Re = 64,000$, $BR=2.87$, Suction = 30 SCFH]
Figure 64. Baseline vs. control boundary layer velocity and turbulence profiles at $x/c \approx 0.37$ at $\alpha = 20^\circ$ at $Re = 64,000$, $BR = 2.87$, Suction = 30 SCFH

Figure 65. Power spectral density of the boundary layer at $x/c \approx 0.37$ and $y/c \approx 0.002$ at $\alpha = 20^\circ$ at $Re = 64,000$, $BR = 2.87$, Suction = 30 SCFH
It appears that the mechanism by which the blowing and suction are effective is as follows. The use of suction is likely pulling the stagnant, separated air towards the suction hole which moves the separation location downstream. The suction is also pulling down on the separated shear layer, which has transitioned to turbulent just downstream of separation. Suction force, if strong enough, pulls the turbulent boundary layer all the way down to the surface where it reattaches and acts like a normal turbulent boundary layer. Once the turbulent boundary layer is reattached, it moves through the adverse pressure gradient and separates where it is predicted that a normal turbulent boundary layer would separate. On the other hand for blowing, leading edge separation is prevented by large three-dimensional structures which are induced by the high blowing ratio jet-cylinder which brings the freestream momentum down to the airfoil surface as turbulent. Again, as with suction, this turbulent boundary layer travels down to the normal location of turbulent separation.

Because the actuation location is slightly downstream of the separation location a significant amount of work is required to change the overall flow. It is noteworthy that when applying suction at $\alpha = 20^\circ$ there is no benefit until 30 SCFH, when there is finally enough suction force to bring the separated shear layer all the way down to the surface. Increasing the suction further, up to the rotameter maximum of 60 SCFH, provided no extra lift increase because the boundary layer is already attached.

3.2.2.5 Summary

Mack et al.\textsuperscript{10} reported that they were unable to reattach the boundary layer from Region IV. This is likely due to the location of their actuation devices and passive trips,
namely at 40% and 30%, respectively. By placing the actuation device at 5% chord, very close to the incidence of laminar separation, passive and active flow control were capable of delaying separation and increasing airfoil performance over a broader range of angles of attack. Conversely, these efforts were unable to delay separation and increase airfoil performance with suction when laminar separation is present a fair distance downstream of the control. Further, at $\alpha = -1^\circ$ zigzag tape is entirely ineffective, again the result of being placed too far upstream of the separation location. These results indicate that a combination of the two configurations could be advantageous; placing active flow control elements at both $x/c \approx 0.05$ and $x/c \approx 0.35$ could provide separation control for all conditions. Further, placing the control closer to and just in front of the separation location should reduce the amount of work required to obtain the lift benefit.

For a brief summary at each angle,

$-1^\circ$: Actuation at 5% chord is too far upstream such that the favorable pressure gradient dampens out any small disturbances created by the zigzag tape and blowing. A moderate blowing ratio needs to be used to create spanwise vortices and disturbances that are large enough to carry downstream to the separated region to transition within the separated shear layer and reattach the flow. This thinned the separated region and increased circulation, therefore lift.

$10^\circ$: The blowing and zigzag tape disturbances have a much shorter distance to travel within the favorable pressure gradient so that smaller amplitudes are sufficient. This allows zigzag tape to be effective, as well
as a low blowing ratio, at improving the velocity profile and inducing transition early in the separated region. Again, the disturbances create transition in the shear layer to reattach the flow and increase circulation and lift.

16°: Here the laminar separation bubble is present near the leading edge and is the optimal choice of flow control. Any method of control only reduced airfoil performance.

20°: Large three-dimensional structures or strong suction force is required in this region to reattach the boundary layer. Blowing induces the large scale structures to move freestream flow down to the surface and suction pulls the boundary layer down. Zigzag tape is completely ineffective.

It is clear that the optimal flow control method, by reviewing Figure 39 and Figure 40, is blowing. It is the only method that produces results in all regions, can be turned off in the laminar separation bubble region, and can be further optimized with closed-loop control. Therefore the remainder of this endeavor includes only the application of blowing as a useful technique.

### 3.3 Case II: \( Re = 1.8 \times 10^5 \)

#### 3.3.1 Baseline

The physics of the airfoil changes considerably after some critical chord Reynolds number is reached, where it demonstrates the lift versus angle of attack and drag shown in Figure 66 and Figure 67, respectively. It is unclear if this is an abrupt change at all
angles of attack or if it occurs gradually. Looking in more detail at Figure 66 and Figure 67, it shows that the experimental data agrees well with XFLR5 in shape, but has slightly lower magnitude for lift. The drag data agrees very well. It is apparent that there is no abrupt formation of a laminar separation bubble, or evidence of premature laminar separation without reattachment. It is suspected that the flow is still initially separating with a laminar boundary layer since this is a laminar airfoil and the Reynolds number is still relatively low compared to its design condition. Therefore, it is suggested that the separation bubble can close on its own at all angle of attack because there is more turbulence in the freestream and the instabilities are amplified from the smaller scales associated with a higher Reynolds number. It is important to note that no well defined regions can be determined, except normal stall.

![Graph showing CL vs. α, Re = 180,000](image)

Figure 66. $C_L$ vs. $\alpha$, $Re = 180,000$, baseline comparison of NACA 643-618
3.3.2 Flow Control

The endeavor of flow control for this application is somewhat different than in Case I because there is no laminar separation without reattachment and the lift curve is not demonstrating sub-par performance. Instead, the goal is to achieve higher lift which means increasing upper surface velocity and to extend the angle of attack range of the airfoil which means delay turbulent separation.

Recall, only blowing is applied to this case study as it is the only control method that demonstrated versatility. The blowing ratio at each angle of attack will remain constant from Case I because the interest is similar flow physics at the sight of the jet. This means quite high jet velocities because not only has the freestream velocity increased, but the $C_p$ has increased as well because the flow is now attached and allows higher circulation and relative upper surface velocity.
Table 4 summarizes the results of the blowing tests at $Re = 180,000$. Upon review, it is clear that flow control at this higher Reynolds number does not provide significant effect. The drag decreases for all angles which is a pleasant benefit, however lift only increases at $\alpha = 16^\circ$ and $20^\circ$. Because extreme levels of blowing are required at $\alpha = 16^\circ$ and $20^\circ$, the figure of merit is very low, thus should only be induced when absolutely necessary. At $10^\circ$ the figure of merit is greater than one, which is because there is more improvement in drag than reduction in lift for a low blowing ratio, so it is considered energy efficient.

A graphical display of these results is given in Figure 68 and Figure 69 for lift and drag, respectively. From these figures, it is obvious that improvement is only truly realized at $\alpha = 16^\circ$ and $20^\circ$, where the blowing provides sufficient additional momentum to extend the angle of attack range and the plateau of maximum lift.

Table 4. Summary of the optimized blowing parameters and performance measures at $Re = 180,000$

<table>
<thead>
<tr>
<th>Region</th>
<th>Angle</th>
<th>Blowing Ratio</th>
<th>Volume Flow (SCFH)</th>
<th>% Lift Increase</th>
<th>% Drag Decrease</th>
<th>Figure of Merit</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>-1°</td>
<td>1.63</td>
<td>60</td>
<td>-22.2</td>
<td>7.0</td>
<td>0.32</td>
</tr>
<tr>
<td>II</td>
<td>10°</td>
<td>0.29</td>
<td>22</td>
<td>-11.5</td>
<td>29.6</td>
<td>1.12</td>
</tr>
<tr>
<td>III</td>
<td>16°</td>
<td>2.02</td>
<td>172</td>
<td>18.8</td>
<td>40.5</td>
<td>0.18</td>
</tr>
<tr>
<td>IV</td>
<td>20°</td>
<td>2.87</td>
<td>255</td>
<td>33.7</td>
<td>19.1</td>
<td>0.15</td>
</tr>
</tbody>
</table>
Figure 68. $C_L$ vs. $\alpha$, $Re = 180,000$, control comparison of NACA 643-618

Figure 69. $C_L$ vs. $C_D$, $Re = 180,000$, control comparison of NACA 643-618
To assist in the determination of the cause of these performance results, surface pressure measurements will again be employed. Here, they compare the blowing upper surface pressure distribution to the baseline and XFLR5 distributions.

Figure 70 displays the results at $\alpha = -1^\circ$, where the baseline case closely follows the predicted XFLR5 viscous and inviscid solution. As Reynolds number increases, the viscous (and therefore baseline) solution should approach the inviscid solution. Note the laminar separation at $x/c \approx 0.5$, which is the same as what is found at $Re = 64,000$. The difference from Case I is that the bubble transitions at $x/c \approx 0.7$, reattaches at $x/c \approx 0.8$, and is without turbulent separation. For reference, it is understood that when examining a pressure distribution, in general, separation occurs when the viscous solution diverges from the inviscid solution. Transition occurs at the sharp pressure increase and reattachment is suggested to occur when the viscous and inviscid solutions re-converge.

Blowing appears to cause the transition to occur sooner, which is similar to what occurred at this angle at $Re = 64,000$. However, here a bubble has already formed in the baseline distribution, and as determined from Region III of Case I (Section 3.2.2.3), applying flow control to a bubble only disrupts its natural ability to improve the flow so the lift actually decreases. Because the bubble size decreases, it provides reason for the decrease in drag.
The pressure distribution of $\alpha = 10^\circ$ can be seen in Figure 71, where the baseline matches the XFLR5 prediction closely, except near the leading edge. It appears as though a separation bubble exists near the leading edge at $x/c \approx 0.1$. Turbulent separation exists at $x/c \approx 0.9$. Blowing, and its associated small vortices, only reduces upper surface velocity because it has again disrupted the laminar separation bubble.
At $\alpha = 16^\circ$ (Figure 72), the trend is similar to that of $10^\circ$, however, the laminar separation bubble is definitely apparent at $x/c \approx 0.05$ in the XFLR5 solution, but is not seen in the baseline due to the smallness of the bubble and the lack of pressure distribution resolution. Turbulent separation exists at $x/c \approx 0.5$. The blowing increases surface velocity a moderate amount through large structure entrainment and delays turbulent separation to $x/c \approx 0.55$ which both increases lift and decreases drag.

The same physics are occurring at $\alpha = 20^\circ$, shown in Figure 73, as at $\alpha = 16^\circ$. The only difference is that the velocity increase and turbulence separation suppression is more significant due to the higher blowing ratio structures.
Figure 72. Pressure distribution comparison of upper surface at $\alpha = 16^\circ$ at $Re = 180,000$, $BR=2.02$

Figure 73. Pressure distribution comparison of upper surface at $\alpha = 20^\circ$ at $Re = 180,000$, $BR=2.87$
3.4 Case III: $Re = 1.0 \times 10^6$

3.4.1 Baseline

A similar investigation has been carried out at the much higher Reynolds number of $Re = 1,000,000$. The results compare well with the predictions of XFLR5 for lift and drag which are respectively presented in Figure 74 and Figure 75. These lift and drag curves trend similarly to Case II, so the expected results are presumed to be similar. The uncertainty bars are not shown here because they are very small. The laminar separation bubble should be present at all angles of attack and turbulent separation becomes important at $a = 10^\circ$. Mayda et al.$^{42}$ suggested that the laminar separation bubble is present up to a Reynolds number of nine million, where beyond that, transition to turbulence is induced before any laminar separation occurs. There is no drastic stall at high angles of attack, which would be evidence of leading edge separation without reattachment. Instead, lift slowly decreases as a result of turbulent separation creeping further forward to disrupt circulation.
Figure 74. $C_L$ vs. $\alpha$, $Re = 1,000,000$, baseline comparison of NACA 643-618

Figure 75. $C_L$ vs. $C_D$, $Re = 1,000,000$, baseline comparison of NACA 643-618
3.4.2 Flow Control

The goal, as in Case II, is to increase overall lift and extend the angle of attack range. The results of blowing actuation are summarized in Table 5, Figure 76 and Figure 77. Recall, the same blowing ratios as the previous two cases are used here to maintain similar control physics. It is readily apparent that blowing at $Re = 1,000,000$ is detrimental to airfoil performance at low ($\alpha = -1^\circ$) and moderate ($\alpha = 10^\circ$) angles of attack. The drag increases significantly and a small lift reduction is noted. At high angles of attack ($\alpha = 16^\circ$ and $20^\circ$), the blowing increases lift and reduces drag, but at a high cost of actuation. A review of the figure of merit shows that blowing is a poor choice of actuation at all angles of attack. It is likely that different blowing ratios would provide improved results, but such a study is not part of the scope of this research.

<table>
<thead>
<tr>
<th>Region</th>
<th>Angle</th>
<th>Blowing Ratio</th>
<th>Volume Flow (SCFH)</th>
<th>% Lift Increase</th>
<th>% Drag Decrease</th>
<th>Figure of Merit</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>-1°</td>
<td>1.63</td>
<td>68</td>
<td>-2.4</td>
<td>-96.9</td>
<td>0.09</td>
</tr>
<tr>
<td>II</td>
<td>10°</td>
<td>0.29</td>
<td>19</td>
<td>-7.1</td>
<td>-37.2</td>
<td>0.38</td>
</tr>
<tr>
<td>III</td>
<td>16°</td>
<td>2.02</td>
<td>133</td>
<td>10.7</td>
<td>15.0</td>
<td>0.35</td>
</tr>
<tr>
<td>IV</td>
<td>20°</td>
<td>2.87</td>
<td>196</td>
<td>19.4</td>
<td>4.1</td>
<td>0.29</td>
</tr>
</tbody>
</table>
To study the cause of the changes in performance, the surface pressure distributions are briefly reviewed. The XFLR5 distribution at $\alpha = -1^\circ$ (Figure 78) shows a laminar
separation bubble at $x/c \approx 0.55$, which cannot be seen in the baseline distribution. This is likely the result of resolution. Blowing has little to no effect on the distribution, a result of the small structures induced, which is corroborated by the small change in lift. The cause of the increase in drag cannot be determined from the pressure distribution and will remain unfounded.

The pressure distribution of $\alpha = 10^\circ$, offered in Figure 79, shows the separation bubble at $x/c \approx 0.05$ which can only be seen in the XFLR5 viscous solution. It is apparent that blowing slightly reduces the velocity on the upper surface giving rise to the lift reduction. Turbulent separation occurs at $x/c \approx 0.75$, which was predicted to become evident at this angle based on the lift curve.

Pressure distributions for $\alpha = 16^\circ$ (Figure 80) and $\alpha = 20^\circ$ (Figure 81) give a similar trend to what $\alpha = 10^\circ$ shows, however the turbulent separation point is moving forward as the angle increases. Blowing actuation delays separation by approximately two tap locations for both cases. Due to the high blowing ratio for each case, the tap at $x/c \approx 0.05$ shows a significant increase in velocity which then drops quickly to a value slightly higher than the baseline case. The velocity increase from momentum entrainment of induced vortices around the high blowing ratio jet improves the boundary layer profile and delays separation in both cases. By skewing or pitching the jet, all of the unused surface-normal momentum addition could be utilized in creating stronger streamwise or spanwise vortices, thereby requiring less mass flow.
Figure 78. Pressure distribution comparison of upper surface at $\alpha = -1^\circ$ at $Re = 1,000,000, BR=1.63$

Figure 79. Pressure distribution comparison of upper surface at $\alpha = 10^\circ$ at $Re = 1,000,000, BR=0.29$
Figure 80. Pressure distribution comparison of upper surface at $\alpha = 16^\circ$ at $Re = 1,000,000$, $BR=2.02$

Figure 81. Pressure distribution comparison of upper surface at $\alpha = 20^\circ$ at $Re = 1,000,000$, $BR=2.87$
3.5 Case IV: $Re = 4.0 \times 10^6$

3.5.1 Baseline

The baseline lift and drag curves, given in Figure 82 and Figure 83, match the trends of Case III very closely. This is because the order of magnitude difference between the two cases is small and the Reynolds numbers are within the suggested operating range for the airfoil, so only minor changes should occur with Reynolds number changes. Again, uncertainty bars are left out because they are small. The experimental data matches well with XFLR5, as expected. The cause for the two peaks of maximum $C_L$ is unknown, but is speculated to be the result of leading edge pressure distribution resolution that does not catch the short suction peak found at these high angles. The onset of turbulent separation appears at $\alpha = 10^\circ$, the same as Case III.

![Figure 82. $C_L$ vs. $\alpha$, $Re = 4,000,000$, baseline comparison of NACA 643-618](image-url)
3.5.2 Flow Control

The review of this section will be brief because it is suggested that the flow physics and performance are the same between this case and Case III, as evidenced by Table 6, Figure 84, and Figure 85. One change is that the apparatus required to maintain a constant blowing ratio from Cases I-III at $\alpha = 16^\circ$ and $20^\circ$ could not be realized, so instead a lower blowing ratio is employed. This provides the same performance enhancement, but at less required power which enables the figure of merit to become greater than one for $\alpha = 20^\circ$. This result strengthens the hypothesis that the blowing amplitude can be adjusted to acquire an optimal figure of merit. Thus, maintaining one blowing ratio throughout all Reynolds numbers is a poor choice.
Table 6. Summary of the optimized blowing parameters and performance measures at $Re = 4,000,000$

<table>
<thead>
<tr>
<th>Region</th>
<th>Angle</th>
<th>Blowing Ratio</th>
<th>Volume Flow (SCFH)</th>
<th>% Lift Increase</th>
<th>% Drag Decrease</th>
<th>Figure of Merit</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>-1°</td>
<td>1.63</td>
<td>189</td>
<td>-3.2</td>
<td>-164.1</td>
<td>0.19</td>
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<tr>
<td>II</td>
<td>10°</td>
<td>0.29</td>
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<td>-1.4</td>
<td>-9.0</td>
<td>0.83</td>
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<tr>
<td>III</td>
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<td>1.60</td>
<td>330</td>
<td>14.4</td>
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<td>IV</td>
<td>20°</td>
<td>1.59</td>
<td>330</td>
<td>20.1</td>
<td>30.6</td>
<td>1.15</td>
</tr>
</tbody>
</table>

Figure 84. $C_L$ vs. $\alpha$, $Re = 4,000,000$, control comparison of NACA 643-618
From the pressure distribution investigation of Figure 86-Figure 89, it is apparent that the distribution follows the XFLR5 and the Case III distribution, so minimal further discussion will be given. The baseline distribution, as predicted earlier, is creeping towards the inviscid solution at Reynolds number increases. Also, as Reynolds number increases, the bubble gets smaller because transition occurs faster in the separated shear layer so it cannot be seen in the distributions. Another interesting point to note is that turbulent separation occurs at approximately the same location here as in Case III.

**Figure 85.** $C_L$ vs. $C_D$, $Re = 4,000,000$, control comparison of NACA 643-618
Figure 86. Pressure distribution comparison of upper surface at $\alpha = -1^\circ$ at $Re = 4,000,000$, $BR=1.63$

Figure 87. Pressure distribution comparison of upper surface at $\alpha = 10^\circ$ at $Re = 4,000,000$, $BR=0.29$
Figure 88. Pressure distribution comparison of upper surface at $\alpha = 16^\circ$ at $Re = 4,000,000$, $BR=1.60$

Figure 89. Pressure distribution comparison of upper surface at $\alpha = 20^\circ$ at $Re = 4,000,000$, $BR=1.59$
3.6 Reynolds Number Scaling

This section is designed to show the relationship between all four Reynolds numbers and the effect of Reynolds number. Discussed first is the lift versus angle of attack curve, shown in Figure 90. An important point to make is that since there is no interest in the effect of compressibility, its effect in this figure has been removed via the Prandtl-Glauert compressibility correction in reverse. Generally, all four curves have similar characteristics, but some differences do exist. When comparing the high to low Reynolds number cases, recall that two different airfoil fabrication techniques are used, two different wind tunnels are used, and two different data acquisition and reduction methods are used.

As suggested from the literature review, it is expected that as Reynolds number increases, so does lift. This trend is apparent in the data, where the lift quickly approaches a limit as the boundary layer thins and separation bubble shrinks, owing to higher lift. The dips in maximum $C_L$ at $\alpha = 12^\circ$ and $14^\circ$ for the high Reynolds number cases are suspected to be a result of a smaller separation bubble which reduces the size of the suction peak therefore decreasing lift.

Blowing delays turbulent separation sufficiently well for all Reynolds numbers, where the percent increase in lift is dependent upon the baseline lift. It is important to note that some increase is always apparent at high angles. This is a consequence of the significant entrainment induced by the large scale vortices.

Blowing at $Re = 64,000$ brings the lift curve up near what is found for the baseline of the rest of the higher Reynolds number cases, which becomes limited by turbulent
separation. Thus, the laminar separation bubble induced by control in Case I is favorable to some other physical induction because it mimics natural characteristics at high Reynolds numbers.

For Cases II-IV, the blowing always reduces lift at $\alpha \leq 10^\circ$ as a result of disruption of the laminar separation bubble through small vortices. For these cases and angles, actuation is unnecessary and should not be employed.

Based on the lift values, it is concluded that actuation should only be used at $Re = 64,000$ or to delay turbulent separation at all Reynolds numbers. No benefit is obtained otherwise.

![Figure 90. $C_L$ vs. $\alpha$, Reynolds numbers comparison of NACA 643-618](image-extension)

As far as drag is concerned, which is presented in Figure 91, drag generally decreases as Reynolds number increases, as expected from the discussion in the literature review.
This is a result of a thinning boundary layer and smaller laminar separation bubble. When blowing is applied below the stall angle, drag generally increases (except for $Re = 64,000$). At above the stall angle, blowing produces a modest drag improvement across all Reynolds numbers.

![Figure 91. $C_L$ vs. $C_D$, Reynolds numbers comparison of NACA 643-618](image)

The figure of merit is plotted against Reynolds number in a log-log plot for the four angles of attack, seen in Figure 92. The dashed connecting lines indicate the tests where blowing ratio was reduced. It is apparent that only three tests had a figure of merit greater than one (shown by the dashed black line), indicating limited energy efficient control. Overall, it can be concluded that efficiency of control decreases with Reynolds number increase at low $\alpha$. It is important to note that normal, steady blowing, in general, is not energy efficient. Instead, pitch and/or skew should be employed to strengthen
particular vortices and increase streamwise momentum. Pulsing should also be investigated, as it is suggested to have the potential to amplify unstable frequencies and also exploit the effect of hysteresis such that reduced mass flow may be required. By employing these enhancements, it may be possible to have energy efficient control over a wider range of Reynolds number and angle of attack.

Figure 92. Trend of figure of merit at each angle of attack as Reynolds number increases
Chapter 4: Conclusions

4.1 Conclusions

Based on the findings above, several key conclusions can be made regarding the relationship between laminar airfoils, separation bubbles, flow control, and Reynolds number. These conclusions are listed as follows,

1) The NACA 643-618 airfoil can be effectively controlled through a range of Reynolds number using simple blowing. The lift curve can be modified to remove any abrupt changes in performance and increase overall performance.

2) Blowing is identified as the best method of the three controls (blowing, suction, and zigzag tape) because it can be controlled, optimized, and shows improvement in all regions. Zigzag tape may be feasible if actuation occurs far enough downstream such that the favorable pressure gradient does not suppress the disturbances. Suction is generally unreliable.

3) Control, in general, should only be employed at very low Reynolds numbers where laminar separation without reattachment is present or to delay turbulent separation at all Reynolds numbers to extend the angle of attack range.
4) Performing minor actuation when the laminar separation bubble is present reduces performance by disrupting the natural shape and location of the bubble.

5) With respect to changing Reynolds numbers, maintaining local blowing ratio is an unfavorable decision if optimizing energy is of great concern. It is recommended that independent studies be carried out at each Reynolds number to determine the most suitable blowing ratio at each.

6) At $Re = 64,000$, low blowing ratios are required at low angles to induce mild entrainment and to ultimately transition the flow earlier. The longer the favorable pressure gradient, the larger amplitude or disturbance required. On the other hand, large blowing ratios are required at high angles of attack to produce large scale structures strong enough to bring a leading edge separation back down to the surface.

7) From an energy conservation standpoint, the figure of merit should be used for any further investigation of airfoil optimization. Since the purpose of a laminar airfoil is to be environmentally friendly, it would be a step backwards to use more energy for actuation to allow the laminar airfoil to operate properly.

8) Based on the control study, a few conclusions can be made regarding blowing parameters. Locating blowing near the leading edge is effective in both inducing vortices to initiate transition and delaying turbulent separation. Low blowing ratios are sufficient when only transition is required and high blowing
ratios are required when large separation is evident. Pulsed blowing would be beneficial, especially when transition is induced, as it is apparent that hysteresis has an important effect. This would allow significantly reduced mass flow with the same performance increase, and could also lock-in and amplify other flow instabilities like the wall-bounded shear layer vortices. Pitch or skew could be well-utilized to generate stronger span- or streamwise vortices and adding momentum to the flow.
References


Appendix A: Wind Tunnel Corrections

The lift coefficient, drag coefficient and angle of attack have all been corrected (denoted subscript “c”) for wind tunnel wall interference for a two-dimensional airfoil, per the equations developed by Barlow et al.\textsuperscript{37} The lift (Equation 13) is corrected for airfoil and wake blockage, and streamline curvature. The drag is corrected similarly in Equation 14. Finally, the angle of attack correction, given in Equation 15, accounts for streamline curvature.

\[
C_{L,c} = C_L(1 - \sigma - 2\varepsilon_{solid} - 2\varepsilon_{wake}) \tag{Equation 13}
\]

\[
C_{D,c} = C_D(1 - 3\varepsilon_{solid} - 2\varepsilon_{wake}) \tag{Equation 14}
\]

\[
\alpha_c = \alpha + \frac{57.3\sigma}{2\pi}
\left( C_L + 4C_M \frac{c}{h} \right) \tag{Equation 15}
\]

The definitions of $\sigma$ and $\varepsilon$ are provided in the following equations, where $h$ is the wind tunnel dimension normal to the airfoil chord (i.e. height). Note that the empirical body shape factor ($A$) is approximately 0.338 for the NACA 643-618. The calculation of
the moment coefficient about the quarter chord ($C_{M,c/4}$) is calculated similarly to lift and drag coefficients from Anderson.²

$$\sigma = \frac{\pi^2}{48} \left(\frac{c}{h}\right)^2$$  \hspace{1cm} \text{Equation 16}

$$\epsilon_{solid} = \Lambda\sigma \quad \epsilon_{wake} = \frac{c}{4h} C_D$$  \hspace{1cm} \text{Equation 17}
Appendix B: Uncertainty Analysis

An uncertainty analysis is performed on the lift coefficients which are shown as error bars on the $C_L$ versus $\alpha$ curve. The calculation of the piecewise lift coefficient using the trapezoidal rule is provided in Equation 18, where $j$ is the piecewise index, $u$ indicates upper surface and $l$ indicates lower surface. To calculate uncertainty, the partial derivatives of Equation 18 are taken with respect to $\alpha$, $cp$, and $x$. These derivates are calculated at each piece and summed across the airfoil. The uncertainty is then obtained by summing each individual variance, shown in Equation 19. The uncertainties of $\alpha$, $cp$, and $x$ are provided in Table 7. A sample of lift coefficient uncertainties is provided in Table 8 to show values typical in this thesis. It is evident that the high speed tunnel has a better setup because the uncertainties are low as a result of well-machined parts and highly accurate pressure transducers.

\[
C_{l,j} = \left( \frac{1}{2} \left( \frac{cp_{j+1} + cp_j}{x_{j+1} - x_j} \right) (x_{j+1} - x_j) - \frac{1}{2} \left( \frac{cp_{j+1} + cp_j}{x_{j+1} - x_j} \right) (x_{j+1} - x_j) \right) \cos(\alpha) - \left( \frac{1}{2} \left( \frac{cp_{j+1} (zu_{j+1} - zu_j)}{x_{j+1} - x_j} \right) (x_{j+1} - x_j) - \frac{1}{2} \left( \frac{cp_{j+1} (zu_{j+1} - zu_j)}{x_{j+1} - x_j} \right) (x_{j+1} - x_j) \right) \sin(\alpha)
\]

Equation 18
$$U_{cL} = \sqrt{\left(\frac{\partial C_L}{\partial \alpha} U_\alpha^2 + \frac{\partial C_L}{\partial x} U_x^2 + \frac{\partial C_L}{\partial C_p} U_{Cp}^2\right)}$$ \hspace{1cm} \text{Equation 19}

Table 7. Uncertainty of independent variables

<table>
<thead>
<tr>
<th>Tunnel</th>
<th>$\alpha$</th>
<th>$x/c$</th>
<th>$C_p$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Low speed</td>
<td>$\pm 0.25$</td>
<td>$\pm 0.003$</td>
<td>$\pm 0.0051$</td>
</tr>
<tr>
<td>High speed</td>
<td>$\pm 0.011$</td>
<td>$\pm 0.00002$</td>
<td>$\pm 0.0005$</td>
</tr>
</tbody>
</table>

Table 8. Uncertainty in $C_L$ at 10°

<table>
<thead>
<tr>
<th>Tunnel</th>
<th>$C_L$</th>
<th>$U_{CL}$</th>
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</thead>
<tbody>
<tr>
<td>Low speed</td>
<td>0.4868</td>
<td>$\pm 0.1148$</td>
</tr>
<tr>
<td>High speed</td>
<td>1.0889</td>
<td>$\pm 0.0013$</td>
</tr>
</tbody>
</table>