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Parametric Flow Visualization of Dynamic Roughness Effects

Vinay Jakkali

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PARAMETRIC FLOW VISUALIZATION OF DYNAMIC ROUGHNESS EFFECTS

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**Thesis submitted to the Benjamin M. Statler College of Engineering and Mineral
Resources**

at West Virginia University

in partial fulfillment of the requirements for the degree of

Master of Science

in

Mechanical Engineering

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Morgantown, West Virginia

2015

Keywords: Flow Control, Flow Visualization, Dynamic Roughness

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ABSTRACT

Parametric Flow Visualization of Dynamic Roughness Effects.

Vinay Jakkali

The ever growing need in the aircraft industry to enhance the performance of a flight vehicle has led to active areas of research which focus on the control of the local boundary layer by both passive and active methods. An effective flow control mechanism can improve the performance of a flight vehicle in various ways, one of which is eliminating boundary layer separation. To be effective the mechanism not only needs to control the boundary layer as desired, but also use less energy than the resulting energy savings. In this study, the effectiveness of an active flow control technique known as dynamic roughness (DR) has been explored to eliminate the laminar separation bubble near the leading edge and also to eliminate the stall on a NACA 0012 airfoil wing.

As opposed to static roughness, dynamic roughness utilizes small time-dependent deforming elements or humps with displacement amplitudes that are on the order of the local boundary layer height to energize the local boundary layer. DR is primarily characterized by the maximum amplitude and operating frequency. A flow visualization study was conducted on a 2D NACA 0012 airfoil model at different angles of attack, and also varying the Reynolds number and DR actuation frequency with fixed maximum DR amplitude. The experimental results from this study suggests that DR is an effective method of reattaching a totally separated boundary layer. In addition, this study discusses some of the fundamental physics behind the working of DR and proposes some non-dimensional terms that may help to explain the driving force behind the mechanism.

Dedication

I would like to dedicate this thesis to my family and my many friends here in the US and back home in India. I would like to extend a special feeling of gratitude to my loving parents, Basavaraj and Pushpa Jakkali, their sacrifices and belief in my abilities led me to reach this point of my life. My brother, Vishal, who has always stood by me and had a huge effect on the man I am today. And last but not the least, I am grateful to my friends, especially, Meenakshi, Rahul, Priyesh, Danielle, Joe and Liz - for treating me as their own family, giving me moral support when I needed it and helping me through one of my life's most difficult stages - the recovery after my injury, but most importantly making a foreign country feel like home.

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Chapter 1

Introduction

In recent times the airfoil aerodynamics at low Reynolds number has gained more attention due to its relevance to micro air vehicles (MAV) and small unmanned air vehicle (UAV) platforms. The size of these vehicles are small relative to traditional aircraft and the slow airspeeds place these vehicles in low Reynolds number region of $10^4 - 10^5$ (1). Also, the flow characteristics at these Reynolds number are very different when compared to flow characteristics at higher Reynolds number.

The parameter C_L/C_D is a good indication of airfoil effectiveness. The fact that at lower Reynolds number the viscous effects are relatively higher causing higher drag and lower lift, the increase in the Reynolds number improves the C_L/C_D value, and as can be seen from Figure 1.1, that at a Reynolds number of about 70,000, the lift-to-drag ratio transitions into higher values thus providing higher performance (2). When compared to high Reynolds number, the low Reynolds number aerodynamics is more susceptible to the presence of leading edge separation of the boundary layer at high angles of attack (3). Many earlier studies conducted by Tani (1), Lissaman (2), Rinoie & Takemura (3), Carmichael (4) and O'Meara & Mueller (5) establish the correlation between the laminar boundary layer flow separation and chord Reynolds number.

The study conducted by Lissaman (2) show the presence of a separation bubble, which occurs when the boundary layer detaches from the airfoil surface and further downstream reattaches as a turbulent boundary layer. Figure 1.2 shows the graphical representation of the laminar separation bubble near the leading edge with downstream reattachment as turbulent boundary layer. The Reynolds number of the flow predominantly influences the size of the separation bubble. Any small disturbance can cause the bubble to burst and cover the upper surface of the airfoil with turbulent boundary layer, causing it to stall.

Figure 1.1: Low Reynolds Number performance transition (2)

Figure 1.2: Leading Edge Separation Bubble (6)

Therefore, it is desirable to improve the performance of an airfoil at high angles of attack by delaying the transition to turbulence or eliminating the leading edge separation. Many different studies have been conducted and different flow control techniques have been employed in the past to energize the boundary layer to delay transition or eliminating the leading edge separation. The flow control techniques can be classified into both passive and active formats. The passive techniques include use of roughness or compliant walls and active methods involve blowing, boundary layer (BL) control (suction on surfaces), circulation control (trailing edge(TE) blown Coanda surfaces), motion of the solid wall, synthetic jets or plasma jets.

One of the active control methods is by the use of the surface mounted deforming elements known as Dynamic Roughness (DR) (7). It utilizes the actuation of one or more surface-mounted deforming elements to energize the fluid within a laminar boundary layer. Earlier studies conducted (6) (7) (8) (9) (10) (11) have indicated that DR is effective in eliminating leading edge laminar separation bubbles at low Reynolds numbers, and is also capable of reattaching a completely separated flow at higher angles of attack.

A study by Huebsch, et. al, (9) looked at the small surface perturbations (now known as DR) to increase the performance of the airfoil at higher angles of attack. Their study included both numerical (2-dimensional and 3-dimensional simulations) and experimental work to show the effectiveness of the DR in eliminating the leading edge separation bubble. They produced an experimental model to validate the numerical simulations.

Figure 1.3 show the relationships between amplitude and frequency with Strouhal number as a function of the scaled roughness with three different regions of flow control, namely, No flow control, Buffer zone, Flow control. It can be seen that there is clear inverse correlation between the amplitudes and the frequencies where flow control is effective. It can be seen that as the amplitude (always sub boundary layer) of the roughness elements is decreased, a higher Strouhal number is needed to obtain flow control. An experimental study conducted by Grager (6), shows that the surface actuation devices have the ability to suppress the leading edge separation and are capable of delaying the stall. This study also showed that the ability to delay stall through the use of these devices increased the lift performance for the lifting surfaces. Also, they found that the higher the operating frequency of the DR, the more likely it is able to prevent the flow separation. The study also shows that the amplitude of the surface roughness elements plays a significant role

in flow control. If the amplitude is higher than that of a local boundary layer thickness, it is significantly less effective in reattaching the boundary layer and if the amplitude is much smaller than that of a local boundary layer, it needs much higher actuation frequencies to achieve reattachment of the boundary layer.

Figure 1.3: Parametric results showing relationship between amplitude and frequency with Strouhal number as a function of the scaled roughness amplitude for three different regions of flow control at $Re=100,000$ (9)

The goal of this study is to assess the effectiveness with the help of flow visualization of DR to eliminate the leading edge separation of flow over NACA 0012 airfoil by varying the chord Reynolds number, DR actuation frequency and the angle of attack.

Chapter 2

Experimental Setup.

The work done is purely an experimental flow visualization effort. Smoke flow visualization was used to investigate flow separation and subsequent flow control. The experimental setup includes the following components.

2.1 Tunnel

The experiment was conducted in a low-turbulence (the turbulence intensity of about 0.75% of the free stream velocity) flow visualization wind tunnel located in the Mechanical and Aerospace Department of West Virginia University. The wind tunnel is capable of wind speeds up to 100 mph and has a test section of 6-inch x 6-inch cross section with transparent walls. Installed in front of the test section are a series of screens with decreasing mesh size followed by a contraction section to help provide uniform, low turbulence flow. The flow visualization tunnel is shown in Figure 2.1.

Figure 2.1: Flow Visualization Wind Tunnel.

2.2 Airfoil Model

The airfoil model tested was a rectangular, untwisted wing with a NACA 0012 airfoil profile with a chord length of 4 inches and a span of 3 inches. The overall size was not only dictated by the test section size, but also the requirement to avoid blockage issues when the airfoil was set to the higher

angles of attack. The blockage at an angle of attack of 16° (the maximum value used in this study) was found to be 10.45%.

The model was designed using SolidWorks 3D CAD software. A baseline airfoil with NACA 0012 profile was printed using a 3D printer, and a mold was prepared by pouring silicon rubber material around the printed model. Polyurethane resin was poured into the mold to cast the model. The curing time for the polyurethane model is about 20 minutes. This method allowed us to quickly make new models as needed.

While casting the model a copper tube of 3/8 inch (as can be seen in Figure 2.2) outer diameter pressed into an elliptical shape was immersed near the leading edge so that it could act as a hollow airtight chamber for the pneumatic actuation of the DR (so that when the model is covered with the latex skin, it would inflate and create tiny bumps when actuated). Figure 2.3 shows the SolidWorks model of the NACA 0012 with the DR section in the leading edge region without the latex or the silicone skin. The back plate and wing tip of the cast model were then machined to the specified dimensions as indicated in Figure 2.4.

Figure 2.2: A polyurethane cast model taken out of mold after the curing.

Figure 2.3: 3D CAD model of non-twisted wing with NACA 0012 profile and also showing DR Region.

Figure 2.4: The cast model being machined to designed specifications

The leading edge DR section has a span of 2 inches (about 50% of the chord length), placed in the middle of the model leaving 0.5 inches towards both the ends without any DR elements. The entire airfoil model was covered (glued to the model using cyano-acrylate glue) with a thin layer of latex rubber (0.012 inches, which proved to be nominal in this case because, anything thinner would stretch and deteriorate faster after being operational and anything thicker needed more power to achieve same amplitude of the roughness elements and was harder to keep it glued to the model; see Figure 2.5), which was painted with chalk board black paint for low-glare flow visualization purposes, and a black silicone skin of thickness 0.006 inches (using a little thicker silicone skin, like 0.012 inches or 0.016 inches, might help in achieving and controlling the amplitude of the

roughness even better) is used on the model which is used for testing angles of attack 13° to 16° . The plane of the holes was tangential to the upper surface near the leading edge and the holes were drilled in such a way that the plane of drilling was normal to the tangential plane to the upper surface near the leading edge. With latex rubber covering the holes, round bumps were formed when the chamber was pressurized, thus creating the tunable dynamic roughness elements. Steel wool was introduced in the hollow airtight copper chamber to eliminate any standing waves from the drive pressure within the chamber and also to get uniform amplitude over the entire DR section when the chamber was pressurized. The air chamber was connected to the actuation system using hollow tube.

The DR elements have a base diameter of 3 mm (about 3% of the chord length) and a measured maximum height of 0.25 mm (about 0.24% of the chord length). The elements are arranged with a 1.5 diameter center-to-center spacing span wise and aligned in the stream wise direction such that the front edge of a row element aligns with the back edge of the row elements in front of it. The DR region starts at 1% of the chord and ends at 10.7% of the chord. The rows of elements are arranged in staggered grid pattern as shown in Figure 2.6.

Figure 2.5: Airfoil model with DR section covered with thin latex skin.

The end plate of the model, which fits into the back plate and is connected to the angle of attack turntable, is a circular plate of diameter 4.5 inches and of 0.146-inch-thick sits in a circular slot of diameter 4.5 inches and 0.146-inch-deep slot on the rectangular back plate of the wind tunnel test section. This set up can be connected to the back wall of the test section by a stud and nut

arrangement. It is easy to vary the angles of attack by loosening the nut and then tightened to the required angle of attack. The back wall of the tunnel has angle graduations on it and there is a reference graduation on the back plate of the model.

Figure 2.6: 3D CAD model showing DR Region with staggered grid pattern.

2.3 Actuation System

The air chamber was pressurized in a time dependent fashion using a modified two cycle, 27 cubic centimeter displacement piston-cylinder drive system connected to a variable speed 0.5 hp electric motor. The electric motor was used to drive the piston-cylinder drive system. A set of pulleys and a belt were used to connect the motor to the engine and increase its maximum output speed. The pressure in the chamber was controlled by varying the relief valve attached to the head of the engine. Figure 2.7 shows the piston-cylinder drive system. Controlling the pressure allowed us to control the maximum displacement amplitude of the hump. A stroboscope was used to measure the pumping frequencies of the engine. And also, the frequency was calculated using the high speed camera. The system was able to produce frequencies up to 90 Hz. The actuation system uses 3.3 kW/m at lowest frequency (10 Hz) and uses 8.98 kW/m at the highest frequency (90 Hz).

Figure 2.7: Piston-Cylinder Drive System.

2.4 Visualization Technique

The flow visualization effort primarily required three components: seed particles that would follow the flow, illumination of the flow and an image capture device.

1. **Seeding:** The flow was seeded with either 1~10 micron olive oil droplets or with Di-Ethyl-Hexyl-Sebacate (DEHS). DEHS was atomized using LaVision's seeder instrument, which uses pressurized air to atomize the oil. Olive oil can be used in place of DEHS to achieve the same results, however use of olive oil produces residue which accumulates over time and has adverse effects of the latex skin.
2. **Illumination:** The flow field was illuminated by a continuous laser beam produced by a Lexel Model 95 Argon Ion Laser. The laser produces two wavelengths of 488 nm and 514.5 nm and requires 1300 mW to 1700 mW to produce the respective laser beams. For our studies 514.5 nm wavelength laser beam was used and the maximum laser beam diameter was about 1.5 mm. The laser sheet was created by reflecting the laser beam using a mirror and converted into a sheet by passing it through a 4 mm diameter glass rod. The sheet was positioned at approximately mid plane (center span of the model).
3. **Image Capturing Device:** Two types of devices were used to collect the data. For capturing the images, a DSLR camera was used with an 18-135 mm standard zoom lens. It was set in the photo burst mode. For collecting a flow video, a Photron FASTCAM SA5 high speed camera was used. It is capable of providing mega pixel resolution at up to 7500 frames per second (fps) and reduced resolution for operation to one million frames per

second, and also with light sensitivity up to ISO 10,000. The camera along with a Nikkor 50 mm lens of an aperture of $f/1.4$ was used to capture the images.

For a test case, 20 images were captured for still images case. The shutter speed was set to $1/4000^{\text{th}}$ of a second and the aperture was kept constant with $f/5.6$. The focal length was 67 mm and with no exposure bias and ISO-6400 for angles of attack 9° and 11° . The shutter speed was set to $1/1000^{\text{th}}$ of a second (reduced the shutter speed due to reduction in the intensity of the laser beam), aperture with $f/5.0$, focal length was 67 mm with no exposure bias and ISO-6400 for angle of attack from 13° to 16° . Approximately 1,500 images were collected continuously to create a high speed video. The images were captured at a frame rate of 4000 frames per second (fps) and a shutter speed of $1/4000^{\text{th}}$ of a second. The lens was kept at a constant aperture with the f stop set at $f-1.4$. The resolution of the images captured was 1024×1024 and ISO >6400 . The images were taken by focusing the camera on the illuminated particles. An image was picked from the video or still images to best represent the observed effect. This was done because, there are some difficulties associated with the smoke, like it is very hard to control and it doesn't pass through the illuminated area all of the times during testing making it even harder to capture the images with good details when compared to dye injection in a water tunnel which is typically much easier to control.

After selecting an image which best represented the observed effect, these were post processed using Adobe Photoshop software mainly to crop the unnecessary pixels with no details, and also, the RGB values of these images were altered by the reducing the amount of blue values and increasing the green values solely for better visual and these values vary for each case depending the RGB graph for the selected image, so they were manually altered to get similar looking images color wise.

Chapter 3

Results and Discussion

3.1 Laminar Separation Bubble

The following discussion provides a detailed study of dynamic roughness effects on the low Reynolds number flow over a NACA0012 wing. The testing was done by varying the chord-length Reynolds number, angle of attack and the different DR actuation frequencies.

Figure 3.1: Angle of attack = 9° , chord Reynolds number = 25,000 and Clean case

At an angle of attack 9° and chord Reynolds number 25,000, the following observations can be made for different DR actuation frequencies. As shown in Figure 3.1, with no DR actuation (Clean case), the presence of laminar separation bubble can be seen clearly near the leading edge. The laminar separation bubble roughly extends up to 32% of the chord length. When the DR is actuated, initially the bubble doesn't disappear. When the DR actuation frequency is set to 10 Hz, as shown in the Figure 3.2, it can be seen that the bubble still exists. Further increasing the DR actuation frequency, the laminar separation bubble doesn't disappear/ burst until the DR actuation frequency of 30 Hz is reached as shown in Figure 3.3. At this frequency the bubble disappears, however, the flow is not completely attached to the wing surface and the boundary layer near the leading edge becomes turbulent. As the DR actuation frequency is further increased the boundary layer becomes less turbulent and is evidently controlled as shown in Figure 3.4. From 50-90 Hz, the boundary layer in the vicinity of DR and downstream remains laminar as shown in Figure 3.5. In Figure 3.5, the DR elements can also be seen near the leading edge. An increase in DR frequency from 50-90

Hz causes no change to the flow and the flow remains attached to the wing, hence controlling the flow and eliminating the stall.

Figure 3.2: Angle of attack = 9° , chord Reynolds number = 25,000 and DR = 10 Hz

Figure 3.3: Angle of attack = 9° , chord Reynolds number = 25,000 and DR = 30 Hz

Figure 3.4: Angle of attack 9° , chord Reynolds number = 25,000 and DR = 50 Hz

Figure 3.5: Angle of attack = 9° , chord Reynolds number = 25,000 and DR = 90 Hz

Similarly, when the chord Reynolds number is increased to 50,000 and the angle of attack is kept constant, the following observations can be made. From Figure 3.6, with no DR actuation we can clearly see a laminar separation bubble near the leading edge. As shown in Figure 3.7, when the DR actuation frequency is set to 10 Hz, it can be seen that the laminar separation bubble does not burst and is not eliminated. As the DR actuation frequency is increased, the laminar separation bubble is not eliminated until the DR actuation frequency of 50 Hz is reached. As shown in Figure 3.8, at the DR actuation frequency of 50 Hz the bubble bursts and the boundary layer near the leading edge becomes turbulent. A further increase in the DR frequency causes the boundary layer to become laminar and the flow reattaches to the wing surface as shown in Figure 3.9. Once again, when the flow control is achieved, the boundary layer is laminar around and downstream of the DR at this Reynolds number.

Figure 3.6: Angle of attack = 9° , chord Reynolds number = 50,000 and Clean case

Figure 3.7: Angle of attack = 9° , chord Reynolds number = 50,000 and $DR = 10$ Hz

Figure 3.8: Angle of attack = 9° , chord Reynolds number = 50,000 and $DR = 50$ Hz

Figure 3.9: Angle of attack 9° , chord Reynolds number 50,000 and $DR = 90$ Hz

The behavior of the boundary layer and the formation of the laminar separation remains the same when the chord Reynolds number is increased to 75,000, 100,000 and 125,000. However, the length and thickness of the laminar separation bubble become smaller with increase in Reynolds number. At chord Reynolds number 75,000, the laminar separation bubble can be seen near the

leading edge for a clean case as shown in Figure 3.10. The laminar separation bubble disappears at DR actuation frequency of 50 Hz and the boundary layer near the DR and downstream becomes turbulent as shown in Figure 3.11. The flow starts to reattach and is controlled when the DR actuation frequency of 70 Hz is reached as shown in Figure 3.12. Further increase in DR actuation frequency causes no change as shown in Figure 3.13.

Figure 3.10: Angle of attack = 9° , chord Reynolds number = 75,000 and Clean case.

Figure 3.11: Angle of attack = 9° , chord Reynolds number = 75,000 and DR = 50 Hz

Figure 3.12: Angle of attack = 9° , chord Reynolds number = 75,000 and DR = 70 Hz

Figure 3.13: Angle of attack = 9° , chord Reynolds number = 75,000 and DR = 90 Hz

At Reynolds number 100,000, a laminar separation bubble can be seen near the leading edge for clean case but is very small compared to previous cases as shown in Figure 3.14. The laminar separation bubble disappears when the DR frequency is 30 Hz and the boundary layer becomes turbulent around and downstream of the DR region as shown in Figure 3.15. The boundary layer remains turbulent until the DR frequency of 50 Hz is reached. Further when the DR frequency is increased to 60 Hz, the boundary layer becomes laminar and the flow starts to reattach to the wing surface as shown in Figure 3.16. A further increase in the DR actuation frequency cause no change as shown in Figure 3.17.

Figure 3.14: Angle of attack = 9° , chord Reynolds number = 100,000 and Clean case

Figure 3.15: Angle of attack = 9° , chord Reynolds number = 100,000 and $DR = 30$ Hz

Figure 3.16: Angle of attack = 9° , chord Reynolds number = 100,000 and $DR = 60$ Hz

Figure 3.17: Angle of attack = 9° , chord Reynolds number = 100,000 and $DR = 90$ Hz

Similarly, at chord Reynolds number 125,000, there is no laminar separation bubble near the leading edge as shown in Figure 3.18. The flow remains laminar near the leading edge and transitions into turbulent flow further downstream. However, any increase in DR actuation frequency has a very little or no effect on the flow as shown in Figure 3.19 and Figure 3.20.

Figure 3.18: Angle of attack = 9° , chord Reynolds number = 125,000 and Clean case

Figure 3.19: Angle of attack = 9° , chord Reynolds number = 125,000 and DR = 50 Hz

Figure 3.20: Angle of attack = 9° , chord Reynolds number = 125,000 and DR = 90 Hz

When the angle of attack is increased to 11° , a laminar separation bubble can be seen near the leading edge for the Clean case similar to the previous test cases. Some similar observations can be made when the wing is set at an angle of attack 11° and a Reynolds number 25,000. As shown in Figure 3.21 with no DR actuation, a laminar separation bubble can be clearly seen near the

leading edge. In this case the laminar separation bubble roughly extends up to 29% of the chord length. Initially the bubble doesn't disappear when the DR is actuated, as the DR frequency is gradually increased, the bubble disappears and the flow gets attached to the wing surface, hence eliminating the stall.

When the DR actuation frequency is set to 10 Hz, the bubble doesn't disappear as shown in Figure 3.22. The bubble doesn't disappear until a DR frequency of 40 Hz is reached. At this frequency the bubble disappears, however the flow is turbulent in the vicinity and downstream of the DR region as shown in Figure 3.23. Further increasing the DR actuation frequency to 80 Hz, the flow around and downstream of the DR region becomes laminar and gets attached to the wing surface as shown in Figure 3.24, hence elimination the stall. A further increase in the DR actuation frequency causes no change and the flow remain attached to the wing surface as shown in Figure 3.25.

Figure 3.21 Angle of attack = 11° , chord Reynolds number = 25,000 and Clean case

Figure 3.22: Angle of attack = 11° , chord Reynolds number = 25,000 and DR = 10 Hz

Figure 3.23: Angle of attack = 11° , chord Reynolds number = 25,000 and DR = 40 Hz

Figure 3.24: Angle of attack = 11° , chord Reynolds number = 25,000 and DR = 80 Hz

Figure 3.25: Angle of attack = 11° , chord Reynolds number = 25,000 and DR = 90 Hz

Similarly, when the chord Reynolds number is increased to 50,000 and the angle of attack is kept constant at 11° , the following observations can be made. From Figure 3.26, with no DR actuation we can clearly see a laminar separation bubble near the leading edge. As shown in Figure 3.27, when the DR actuation frequency is set to 10 Hz, it can be seen that the laminar separation bubble

does not disappear and is not eliminated. As the DR actuation frequency is increased, the laminar separation bubble is not eliminated until the DR actuation frequency of 40 Hz is reached. As shown in Figure 3.28, at the DR actuation frequency of 40 Hz the bubble disappears and the boundary layer near the leading edge becomes turbulent. However, the boundary layer becomes comparatively less turbulent with the increase in the DR frequency. The boundary layer near the leading edge becomes laminar when the DR actuation frequency of 70 Hz is reached as shown in Figure 3.29. Beyond the DR actuation frequency of 70 Hz, the boundary layer stays laminar and the flow remains attached to the wing surface as shown in Figure 3.30.

Figure 3.26: Angle of attack = 11° , chord Reynolds number = 50,000 and Clean case

Figure 3.27: Angle of attack = 11° , chord Reynolds number = 50,000 and DR = 10 Hz

Figure 3.28: Angle of attack = 11° , chord Reynolds number = 50,000 and $DR = 40$ Hz

Figure 3.29: Angle of attack = 11° , chord Reynolds number = 50,000 and $DR = 70$ Hz

Figure 3.30: Angle of attack = 11° , chord Reynolds number = 50,000 and $DR = 90$ Hz

The behavior of the boundary layer and the formation of the laminar separation remains the same when the chord Reynolds number is increased to 75,000, 100,000 and 125,000. However, the length and thickness of the laminar separation bubble becomes smaller with increase in Reynolds number. At chord Reynolds number 75,000, the laminar separation bubble can be seen near the

leading edge for a clean case as shown in Figure 3.31. The laminar separation bubble disappears at DR actuation frequency of 50 Hz and the boundary layer near the DR and downstream becomes turbulent as shown in Figure 3.32. The flow starts to reattach and is controlled when the DR actuation frequency of 70 Hz is reached as shown in Figure 3.33. Further increase in DR actuation frequency causes no change as shown in Figure 3.34.

Figure 3.31: Angle of attack = 11° , chord Reynolds number = 75,000 and Clean case

Figure 3.32: Angle of attack = 11° , chord Reynolds number = 75,000 and DR = 50 Hz

Figure 3.33: Angle of attack = 11° , chord Reynolds number = 75,000 and DR = 70 Hz

Figure 3.34: Angle of attack = 11° , chord Reynolds number = 75,000 and DR = 90 Hz

At Reynolds number 100,000, a laminar separation bubble can be seen near the leading edge for clean case but is very small compared to previous cases as shown in Figure 3.35. The laminar separation bubble disappears when the DR frequency is 40 Hz and the boundary layer becomes turbulent around and downstream of the DR region as shown in Figure 3.36. The boundary layer remains turbulent until the DR frequency of 60 Hz is reached. Further when the DR frequency is increased to 70 Hz, the boundary layer becomes laminar and the flow starts to reattach to the wing surface as shown in Figure 3.37. A further increase in the DR actuation frequency cause no change as shown in Figure 3.38.

Figure 3.35: Angle of attack = 11° , chord Reynolds number = 100,000 and clean case

Figure 3.36: Angle of attack = 11° , chord Reynolds number = 100,000 and DR = 40 Hz

Figure 3.37: Angle of attack = 11° , chord Reynolds number = 100,000 and DR = 70 Hz

Figure 3.38: Angle of attack = 11° , chord Reynolds number = 100,000 and DR = 90 Hz

Similarly at chord Reynolds number 125,000, there is no laminar separation bubble near the leading edge as shown in Figure 3.39. The flow remains laminar near the leading edge and transitions into turbulent flow further downstream. However, any increase in DR actuation frequency has a very little or no effect on the flow as shown in Figure 3.40 and Figure 3.41.

Figure 3.39: Angle of attack = 11° , chord Reynolds number = 125,000 and Clean case

Figure 3.40: Angle of attack = 11° , chord Reynolds number = 125,000 and DR = 50 Hz

Figure 3.41: Angle of attack = 11° , chord Reynolds number = 125,000 and DR = 90 Hz

The Figure 3.42 show a summary of the flow visualization images of the clean case at different Reynolds number and at angles of attack 9° and 11° . From the figure, the presence of a laminar separation bubble near the leading edge can be seen. From the image for $\alpha = 9^\circ$ / $Re = 25,000$, the laminar separation bubble roughly extends up to $32 \pm 3\%$ of the chord length. As the Reynolds number increases, the chord wise size of the bubble decreases, i.e, the bubble size at 9° / $Re = 50,000$ is roughly about $21 \pm 3\%$ and it decreases up to less than $10 \pm 3\%$ at $Re = 100,000$ and disappears at the chord Reynolds number 125,000. Similar behavior can be seen when the angle of attack is increased to 11° . The bubble length decreases as the Reynolds number increases. As suggested in the earlier work, the bubble size is predominantly influenced by the Reynolds number of the flow. Therefore, in general the bubble length is inversely dependent on the Reynolds number of the flow. However, the bubble thickness is proportional to its length. The point of flow separation is based on the angle of attack but not the Reynolds number. The graphical representation of the percentage

bubble length chord wise at different Reynolds number at both angles of attack 9° and 11° is shown in Figure 3.43

Figure 3.42: Flow Visualization images showing clean cases at angle of attack 9° and 11° and at different Reynolds number

Figure 3.43: Chord Reynolds Number vs Percentage Bubble Length

As we can see from the images before that the roughness elements in the last row are bigger when compared to the other rows, mainly because of the deterioration of the latex skin. However, even with the bigger size of the roughness elements in the last row, the DR does not affect the flow in a major way, like making the BL more turbulent and it doesn't appear to be greater than the oncoming BL. It is further evident that the majority of the work of inducing the energy to the flow is done by the first row of the DR and also, the placement of the first row of the DR elements. Figure 3.44 shows the wing at 11° and Reynolds number 125,000 Clean case and 90 Hz case. We can see that there is no difference between the two images, i.e, the last element of the DR does not affect the flow.

Figure 3.44: $AoA = 11^\circ$, $Re = 125,000$. Clean case and 90 Hz.

3.2 Effectiveness of Dynamic Roughness

An extensive parametric study of Dynamic Roughness flow control is investigated by varying angle of attack from 13° to 16° (restricted by tunnel blockage limitation) in the increment of 1° and also varying the chord Reynolds number from 25,000 to 125,000 in the increments of 25,000. For each degree of angle of attack and each Reynolds number case, the DR frequency was varied from 0 Hz (Clean case) to 90 Hz in the increments of 10 Hz.

When the wing is set at an angle of attack 13° or higher, the flow starts to separate from the leading edge causing the wing to stall. The separated shear layer at the leading edge transitions into turbulent structures further downstream. When the DR is actuated, initially there is no effect on the boundary layer. Once the DR frequency reaches a certain frequency, in each case the flow starts to reattach. This frequency is known as the threshold frequency. The threshold frequency is different for each angle of attack and Reynolds number. Further increase in the DR actuation frequency beyond the threshold frequency causes the flow to reattach to the wing surface likely eliminating the stall.

Figure 3.45 represents the flow visualization images at angle of attack equal to 13° , Reynolds number 25,000 and at different DR frequencies. As seen from Figure 3.45 (a), the flow starts to separate from the leading edge causing the wing to stall. When the DR frequency is set to 10 Hz, there is no change in flow behavior as shown in Figure 3.45 (b). Once the DR frequency is increased to 30 Hz the flow starts to reattach. However, the flow has not completely reattached to the wing surface as shown in Figure 3.45 (c). For further increases in the DR frequency up to 60 Hz, the flow shows an intermediate effect that is, the flow is neither completely separated nor it is completely reattached. When the DR frequency is set to 60 Hz, we can see that the flow has completely reattached to the surface of the wing, likely eliminating the stall as shown in Figure 3.45 (d). Further increase in the DR frequency to 90 Hz causes very little or no change to the reattached flow as shown in Figure 3.45 (e).

Similarly, Figure 3.46 represents the flow visualization images at angle of attack equal to 13° , Reynolds number 50,000 and different DR frequencies. As seen from Figure 3.46 (a), the flow is separated at the leading edge. When the DR frequency is set to 10 Hz, there is no change in flow as shown in Figure 3.46 (b). Further when the DR frequency is increased to 40 Hz the flow starts

to reattach. However, the flow has not completely reattached to the wing surface, showing an intermediate control as shown in Figure 3.46 (c). This intermediate effect is seen until the DR frequency of 70 Hz is reached. When the DR frequency is set to 70 Hz, it is seen that the flow has completely reattached to the surface of the wing, likely eliminating the stall as shown in Figure 3.46 (d). Further increase in the DR frequency to 90 Hz cause very little or no change to the reattached flow as shown in Figure 3.46 (e).

At this angle of attack, when the Reynolds number is set at 75,000 or higher we don't see much of a leading edge separation. There is no change in the flow, even when the DR frequency is increased to 90 Hz. Figure 3.47 (a) and (b) show the wing at 13° , Reynolds number 75,000 for the clean case and at the DR frequency 90 Hz respectively. As we can see from the Figure 3.47 there is no change in the flow. A similar behavior can be seen when the Reynolds number is increased to 100,000 and 125,000. Figure 3.48 (a) and (b) represent the wing at angle of attack 13° , Reynolds number 100,000 for the clean case and at the DR frequency 90 Hz respectively. Figure 3.49 (a) and (b) represent the wing at angle of attack 13° , Reynolds number 125,000 for the clean case and at the DR frequency 90 Hz respectively.

Figure 3.45: $AoA = 13^\circ$, $Re = 25,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 60 Hz and (e) 90 Hz.

Figure 3.46: $AoA = 13^\circ$, $Re = 50,000$. (a) Clean, (b) 10 Hz, (c) 40 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.47: $AoA = 13^\circ$, $Re = 75,000$. (a) Clean case and (b) 90 Hz.

Figure 3.48: $AoA = 13^\circ$, $Re = 100,000$. (a) Clean case and (b) 90 Hz.

Figure 3.49: $AoA = 13^\circ$, $Re = 125,000$. (a) Clean case and (b) 90 Hz.

Now the results for all the test cases, have been tabulated for when the wing is set at different angles of attack of 13° , 14° , 15° and 16° . At a given angle of attack and Reynolds number, the DR frequency is increased from 0 to 90 Hz. For a completely separated flow near the leading edge, we will list the effect as “No control” and assign a value of “0” in the table. When the flow starts to reattach but is not completely reattached, (i.e, it is neither completely separated at leading edge nor is completely reattached to the wing surface), we will list this effect as “Intermediate control” and assign a value of “0.5” in the table. Further when the flow is completely reattached to the surface, it will be listed as “Complete control” and assigned a value of “1” in the table. Further when there is no change in the flow, for example, in case of 13° , and Reynolds number 75,000 or higher we will list it as “No Change” and include NA in the table for these cases.

The tabulated results and a graphical representation of the tabulated results of a test case will be followed by the respective figures showing the results, except for 13° angle of attack case which has already been presented. All the tabulated results can be compared with the figures shown in Appendix A (it has all the images collected from angles of attack 9° to 16°). For angle of attack 13° the figures from Figure A. 11 to Figure A. 15, and for angle of attack 14° the figures are from Figure A. 16 to Figure A. 20. Table 3.1 and Table 3.2 show the results at different cases when the airfoil is set at angle of attack 13° and 14° respectively and Figure 3.50 and Figure 3.51 show the graphical representation of these three control regions at angles of attack 13° and 14° respectively. Figure 3.52 to Figure 3.56 represent the flow visualization images at Reynolds number 25,000, 50,000, 75,000, 100,000 and 125,000 respectively. However, the figures will only show, a clean case (Figure (a) in all the following figures), a 10 Hz case (Figure (b) in all the following figures), a figure showing intermediate effect (Figure (c) in all the following figures), a figure showing the reattached flow (Figure (d) in all the following figures), and a 90 Hz case showing no further change from figure (d) (Figure (e) in all the following figures).

For angle of attack 14° , when the Reynolds number is 100,000 and 125,000, very little leading edge separation is seen, and there is no change in the flow even when the DR is actuated. These cases are similar to the 13° , 75,000, 100,000 and 125,000 cases. Hence Figure 3.55 and Figure 3.56 will only show the clean case and a 90 Hz case.

Table 3.1: Results at angle of attack 13°

Re \ DR freq	25,000	50,000	75,000	100,000	125,000
Clean	0	0	NA	NA	NA
10 Hz	0	0	NA	NA	NA
20 Hz	0	0	NA	NA	NA
30 Hz	0.5	0	NA	NA	NA
40 Hz	0.5	0.5	NA	NA	NA
50 Hz	0.5	0.5	NA	NA	NA
60 Hz	1	0.5	NA	NA	NA
70 Hz	1	1	NA	NA	NA
80 Hz	1	1	NA	NA	NA
90 Hz	1	1	NA	NA	NA

Figure 3.50: Graphical Representation of tabulated results at 13°

Table 3.2: Results at angle of attack 14°

Re \ DR freq	25,000	50,000	75,000	100,000	125,000
Clean	0	0	0	NA	NA
10 Hz	0	0	0	NA	NA
20 Hz	0	0	0	NA	NA
30 Hz	0.5	0.5	0.5	NA	NA
40 Hz	0.5	0.5	0.5	NA	NA
50 Hz	0.5	0.5	0.5	NA	NA
60 Hz	0.5	0.5	1	NA	NA
70 Hz	1	1	1	NA	NA
80 Hz	1	1	1	NA	NA
90 Hz	1	1	1	NA	NA

Figure 3.51: Graphical Representation of tabulated results at 14°

Figure 3.52: $AoA = 14^\circ$, $Re = 25,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.53: $AoA = 14^\circ$, $Re = 50,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.54: $AoA = 14^\circ$, $Re = 75,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 60 Hz and (e) 90 Hz.

Figure 3.55: $AoA = 14^\circ$, $Re = 100,000$. (a) Clean case and (b) 90 Hz.

Figure 3.56: $AoA = 14^\circ$, $Re = 125,000$. (a) Clean case and (b) 90 Hz.

The Table 3.3 shows the results and Figure 3.57 show the graphical representation for the case when the airfoil was set at angle of attack 15° . The following Figure 3.58 to Figure 3.62 represent the flow visualization images of the same case at Reynolds number 25,000, 50,000, 75,000, 100,000 and 125,000 respectively. For all the results refer figures from Figure A. 21 to Figure A. 25.

Table 3.3: Results at angle of attack 15°

Re \ DR freq	25,000	50,000	75,000	100,000	125,000
Clean	0	0	0	0	0
10 Hz	0	0	0	0	0
20 Hz	0	0	0	0	0.5
30 Hz	0.5	0.5	0.5	0.5	0.5
40 Hz	0.5	0.5	0.5	0.5	1
50 Hz	0.5	0.5	1	1	1
60 Hz	1	0.5	1	1	1
70 Hz	1	1	1	1	1
80 Hz	1	1	1	1	1
90 Hz	1	1	1	1	1

Figure 3.57: Graphical Representation of tabulated results at 15°

Figure 3.58: $AoA = 15^\circ$, $Re = 25,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 60 Hz and (e) 90 Hz.

Figure 3.59: $AoA = 15^\circ$, $Re = 50,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.60: $AoA = 15^\circ$, $Re = 75,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 50 Hz and (e) 90 Hz.

Figure 3.61: $AoA = 15^\circ$, $Re = 100,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 50 Hz and (e) 90 Hz.

Figure 3.62: $AoA = 15^\circ$, $Re = 125,000$. (a) Clean, (b) 10 Hz, (c) 20 Hz, (d) 40 Hz and (e) 90 Hz.

The Table 3.4 show the results and Figure 3.63 show the graphical representation for the case when the airfoil was set at angle of attack 16° . The following Figure 3.64 to Figure 3.68 represent the flow visualization images of the same case at Reynolds number 25,000, 50,000, 75,000, 100,000 and 125,000 respectively. For all the results refer figures from Figure A. 26 to Figure A. 30

Table 3.4: Results at angle of attack 16°

Re \ DR freq	25,000	50,000	75,000	100,000	125,000
Clean	0	0	0	0	0
10 Hz	0	0	0	0	0
20 Hz	0	0	0	0	0
30 Hz	0.5	0	0	0	0
40 Hz	0.5	0.5	0.5	0.5	0
50 Hz	0.5	0.5	0.5	0.5	0.5
60 Hz	0.5	0.5	0.5	0.5	0.5
70 Hz	1	1	1	1	0.5
80 Hz	1	1	1	1	1
90 Hz	1	1	1	1	1

Figure 3.63: Graphical Representation of tabulated results at 16°

Figure 3.64: $AoA = 16^\circ$, $Re = 25,000$. (a) Clean, (b) 10 Hz, (c) 30 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.65: $AoA = 16^\circ$, $Re = 50,000$. (a) Clean, (b) 10 Hz, (c) 40 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.66: $AoA = 16^\circ$, $Re = 75,000$. (a) Clean, (b) 10 Hz, (c) 40 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.67: $AoA = 16^\circ$, $Re = 100,000$. (a) Clean, (b) 10 Hz, (c) 40 Hz, (d) 70 Hz and (e) 90 Hz.

Figure 3.68: $AoA = 16^\circ$, $Re = 125,000$. (a) Clean, (b) 10 Hz, (c) 50 Hz, (d) 80 Hz and (e) 90 Hz.

3.3 DR Strength Number

Now considering the governing equations for a 2D laminar boundary layer (although, the model uses 3D roughness elements, for the sake of the defining the dimensionless terms and following calculations, we will consider 2D flow) are given by:

Mass:

$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0 \quad (1)$$

x – momentum:

$$\frac{\partial u}{\partial t} + u \frac{\partial u}{\partial x} + v \frac{\partial u}{\partial y} = -\frac{1}{\rho} \frac{dp}{dx} + \nu \frac{\partial^2 u}{\partial y^2} \quad (2)$$

y– momentum:

$$\frac{\partial p}{\partial y} = 0 \quad (3)$$

3.3.1 Non-dimensionalization of x – momentum:

Length: $x^* = \frac{x}{L}$; $y^* = \frac{y}{\delta}$ Where L is chord length and δ is Boundary layer thickness.

Time: $t^* = tf$ Where f is DR frequency.

Velocity: $u^* = \frac{u}{U}$; $v^* = \frac{v}{fA}$ Where U is free stream velocity, A is amplitude of roughness and fA is DR velocity scale.

Pressure: $p^* = \frac{p}{\rho U^2}$:

3.3.2 Substituting in x – momentum:

$$\left(\frac{fL}{U}\right) \frac{\partial u^*}{\partial t^*} + u^* \frac{\partial u^*}{\partial x^*} + \left(\frac{fL}{U}\right) \left(\frac{A}{\delta}\right) v^* \frac{\partial u^*}{\partial y^*} = -\frac{dp^*}{dx^*} + \frac{1}{Re} \left(\frac{L}{\delta}\right)^2 \frac{\partial^2 u^*}{\partial y^{*2}} \quad (4)$$

Defining the following terms:

Strouhal number:

$$F_{DR} = \frac{fl}{U} = \frac{f}{\left(\frac{U}{L}\right)} = \frac{DR \text{ frequency}}{Flow \text{ Frequency}} \quad (5)$$

DR amplitude number:

$$A_{DR} = \frac{A}{\delta} = \frac{DR \text{ amplitude}}{Boundary \text{ layer thickness}} \quad (6)$$

DR Strength number:

$$S_{DR} = F_{DR} * A_{DR} = \left(\frac{fL}{U}\right) \left(\frac{A}{\delta}\right) = \frac{fA}{U\left(\frac{\delta}{L}\right)} = \frac{DR \text{ velocity scale}}{Normal \text{ velocity scale}} \quad (7)$$

Now calculating the boundary layer thickness using Blasius solution for flat plate. In this study the x value i.e, the distance is calculated by measuring the distance between the stagnation point at a given angle of attack and the starting of the DR region (which is located at 1% of the chord length). The stagnation point is extracted from XFLR5 simulations for NACA 0012 airfoil (for $\alpha = 16^\circ$, it came out to be 0.012L, where L is the chord length). After calculating δ at the starting of DR region for different Reynolds number. After calculating the DR strength number for different Reynolds number and plotting it against the Strouhal number we get the plot shown in Figure 3.69.

Figure 3.69: Strouhal number vs DR Strength Number

From Figure 3.69 it can be speculated that, for a given angle of attack as the Reynolds number increases, smaller DR strength number is required to achieve the complete control over the flow. From the tabulated results and from the graph shown in Figure 3.69, it can be said that the DR starts affecting the flow when the DR strength number is ≥ 1 . However, for the case mentioned in this study, the DR starts affecting the flow once a DR strength number is > 3 is reached for Reynolds number 25000. This is mainly because, the boundary layer thickness was calculated using Blasius Solution for the flat plate. It can be safely said that measuring the BL thickness is required to accurately calculate the DR strength number.

Chapter 4

Conclusions

A flow visualization study was conducted to map the efficacy of the DR at different angles of attack and different Reynolds number on a non-twisted NACA 0012 wing fitted with end plates. The study shows that the DR is effective in suppressing the formation of leading edge separation bubble at low Reynolds numbers and at angles of attack 9° and 11° . The DR has been an effective active control method capable of delaying the stall as seen in the cases of angles of attack 13° to 16° . This ability to delay the stall further increases the lift performance of the wing.

The DR tends to be more effective in the cases of lower Reynolds numbers i.e, 25,000 and 50,000 in case of 13° angle of attack. For the rest of the Reynolds number cases at this AoA, i.e, 75,000, 100,000 and 125,000, the flow is not at all separated at the leading edge and the actuation of DR has minimal effect on the flow. And in case of 14° angle of attack, DR is effective in cases of Reynolds numbers of 25,000, 50,000 and 75,000. The flow is not at all separated at the leading edge for Reynolds numbers of 100,000 and 125,000; hence the minimal effect of DR on the flow in this case. Again, the testing was limited to $\text{AoA} \leq 16^\circ$ due to the wind tunnel test section blockage considerations; here the blockage was 10.45% at $\text{AoA} = 16^\circ$.

From the studies, it can be seen that the flow remains attached to the wing surface at higher actuation frequencies. That is, the higher the frequency of the actuation devices, the greater the likeliness of the prevention of separated flow near the leading edge. It is also found that the amplitude of the roughness that are a fraction of boundary layer height are much more effective both in suppressing the leading edge laminar separation bubble and in reattaching the boundary layer. The amplitude of the roughness elements larger than the boundary layer has much less effect in reattaching the boundary layer. The much smaller displacements heights require higher actuation frequencies to achieve the same result.

In this study we also have attempted to explain the theory behind the working of the DR. A few non dimensional numbers like Strouhal number, DR amplitude number and DR strength number were defined and it has been established that, for the DR to be effective, the DR strength number has to be greater than 1.

Further studies are required to investigate the feasibility of the DR actuation. There are only few studies which provide the proof of increased lift for DR. This increased lift needs to be compared with power consumed by the actuation devices, and also a study of the change in the drag forces at these higher angles of attack needs to be done. A further investigation needs to be performed to prove the theory behind the working of DR, i.e, by measuring the boundary layer thickness in each of the test cases by either using Particle Image Velocimetry or Laser Doppler Velocimetry or Hot Wire Anemometry and also, making the amplitude of the roughness elements much smaller and achieving comparatively higher frequencies for these roughness elements. Also recommending flow visualization for same airfoil and Reynolds numbers ranges at even higher angles of attack to get the stall at the highest Reynolds number value for clean cases. It would also be an interesting study to use a macro lens to investigate the laminar separation bubble region.

Although, the DR proved to be an effective method, there were few challenges associated with it. Few of the challenges faced during testing for this study are, the latex skin deteriorated faster, it would lose its strength causing it act as a static roughness which in turn affected the flow, hence was required to replace it again and again. The size of the model used for testing was very small, which was comparatively weaker to handle the pressure generated by the actuation system. So, it would be better to use a bigger model and thicker skin and any different material for skin which doesn't either deteriorate fast or lose its strength might give better control over the testing and also, might be helpful in getting comparatively better results.

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Appendix A: Additional Test Results.

Figure A. 1: Angle of attack = 9° , chord Reynolds number = 25,000 and different DR frequencies.

Figure A. 2: Angle of attack = 9° , chord Reynolds number = 50,000 and different DR frequencies.

Figure A. 3: Angle of attack = 9° , chord Reynolds number = 75,000 and different DR frequencies.

Figure A. 4: Angle of attack = 9° , chord Reynolds number = 100,000 and different DR frequencies.

Figure A. 5: Angle of attack = 9° , chord Reynolds number = 125,000 and different DR frequencies.

Figure A. 6: Angle of attack = 11° , chord Reynolds number = 25,000 and different DR frequencies.

Figure A. 7: Angle of attack = 11° , chord Reynolds number = 50,000 and different DR frequencies.

Figure A. 8: Angle of attack = 11° , chord Reynolds number = 75,000 and different DR frequencies.

Figure A. 9: Angle of attack = 11° , chord Reynolds number = 100,000 and different DR frequencies.

Figure A. 10: Angle of attack = 11° , chord Reynolds number = 125,000 and different DR frequencies.

Figure A. 11: Angle of attack = 13° , chord Reynolds number = 25,000 and different DR frequencies.

Figure A. 12: Angle of attack = 13° , chord Reynolds number = 50,000 and different DR frequencies.

Figure A. 13: Angle of attack = 13° , chord Reynolds number = 75,000 and different DR frequencies.

Figure A. 14: Angle of attack = 13° , chord Reynolds number = 100,000 and different DR frequencies.

Figure A. 15: Angle of attack = 13° , chord Reynolds number = 125,000 and different DR frequencies.

Figure A. 16: Angle of attack = 14° , chord Reynolds number = 25,000 and different DR frequencies.

Figure A. 17: Angle of attack = 14° , chord Reynolds number = 50,000 and different DR frequencies.

Figure A. 18: Angle of attack = 14° , chord Reynolds number = 75,000 and different DR frequencies.

Figure A. 19: Angle of attack = 14° , chord Reynolds number = 100,000 and different DR frequencies.

Figure A. 20: Angle of attack = 14° , chord Reynolds number = 125,000 and different DR frequencies.

Figure A. 21: Angle of attack = 15° , chord Reynolds number = 25,000 and different DR frequencies.

Figure A. 22: Angle of attack = 15° , chord Reynolds number = 50,000 and different DR frequencies.

Figure A. 23: Angle of attack = 15° , chord Reynolds number = 75,000 and different DR frequencies.

Figure A. 24: Angle of attack = 15° , chord Reynolds number = 100,000 and different DR frequencies.

Figure A. 25: Angle of attack = 15° , chord Reynolds number = 125,000 and different DR frequencies.

Figure A. 26: Angle of attack = 16° , chord Reynolds number = 25,000 and different DR frequencies.

Figure A. 27: Angle of attack = 16° , chord Reynolds number = 50,000 and different DR frequencies.

Figure A. 28: Angle of attack = 16° , chord Reynolds number = 75,000 and different DR frequencies.

Figure A. 29: Angle of attack = 16° , chord Reynolds number = 100,000 and different DR frequencies.

Figure A. 30: Angle of attack = 16° , chord Reynolds number = 125,000 and different DR frequencies.

Appendix B: Percentage Blockage Calculation

The tunnel blockage percentage can be calculated using Eqn (8):

$$\%Blockage = \frac{Frontal\ Area}{Cross - sectional\ area\ of\ test\ section} * (100) \quad (8)$$

In our case the cross-sectional area of the test section is 36 in². The frontal area (A_f) is the sum of the frontal area of the wing and the frontal area of the end plate. In this case the frontal area of the end plate is 0.42 in², which remains constant throughout.

The frontal area of the NACA 0012 wing model can be calculated through the following steps. The chord length of the wing is 4 inches; based on this, the maximum thickness of the NACA 0012 wing can be calculated as 12% of the chord length, i.e, 0.48 inches. An unmodified NACA 0012 (or any NACA 4 digit airfoil) wing has the maximum thickness located at 30% of the chord length from the leading edge. In this case it is 1.2 inches from the leading edge. The Figure B. 1 below shows an airfoil at an angle of attack α , with the marked maximum thickness location i.e, 30% of the chord length (0.3c) from the leading edge and half the thickness of the model on the upper side of the wing at this point is 0.06c, and is also shown in the figure, where 'c' represents the chord length of the model.

Figure B. 1: Wing at any angle of attack α (12)

At any angle of attack α , the component of the half thickness on the upper half of the model in y-direction is shown as the red dotted line in the Figure B. 2, which is equal to the y-component of the 30% chord length, which is shown as a blue dotted line in the Figure B. 2.

Figure B. 2: Zoomed in view of the leading edge. (12)

At any angle of attack α , both of these y components are equal and have the same frontal area. Now calculating this angle of attack α , when these y components are equal to each other and is given by Eqn (9).

$$\alpha = \tan^{-1}\left(\frac{0.06c}{0.3c}\right) \quad (9)$$

After calculating, for this model, the $\alpha = 11.3^\circ$, which means that for any α value above this $\alpha = 11.3^\circ$, the leading edge is above the maximum thickest part of the model. Therefore, now the frontal area of the wing can be calculated as the sum of the y component of the chord length and the y component of the leading edge radius above the leading edge starting point, and this whole quantity multiplied by the span of the model. Figure B. 3 shows the zoomed in part of the leading edge.

Figure B. 3: Leading edge circle of radius R_{le} (12)

The leading edge radius is calculated using Eqn (10) (13).

$$R_{le} = 0.0158c \quad (10)$$

Therefore, the part of the y component above the starting point of the leading edge can be calculated using Eqn (11).

$$R_{le}^y = 0.0158c - 0.0158c(\tan \alpha) \quad (11)$$

So for this model, at a maximum angle of attack of 16° , the $R_{le}^y = 0.0112$ inches, and the y component of the chord length at this angle of attack can be calculated using Eqn (12).

$$c_y = c \sin \alpha \quad (12)$$

which comes out to be 1.1025 inches. Now adding both these results and multiplying it with the span of the model (which is 3 inches), the frontal area of the wing at 16° is found to be 3.3411 in^2 . Now to calculate the total frontal area A_f , adding the frontal area of the end plate and the frontal area of the wing at 16° , which comes out to be 3.7611 in^2 . Now using Eqn (8), calculating the % Blockage of the test section, which comes out to be 10.45% for an angle of attack 16° .