



**REPUBLIC OF TÜRKİYE
ADANA ALPARSLAN TÜRKER SCIENCE AND TECHNOLOGY
UNIVERSITY**

**GRADUATE SCHOOL
AEROSPACE ENGINEERING DEPARTMENT**

**THE EFFECT OF LEADING EDGE MORPHING ON THE
AERODYNAMIC CHARACTERISTICS OF NACA 0012 AIRFOIL AT
LOW REYNOLDS NUMBER**

TOLGA BARAN OZDEMIROGLU

M.Sc.



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**THESIS ADVISOR
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ADANA, 2025

ÖZET

DÜŞÜK REYNOLDS SAYISINDA NACA 0012 KANAT PROFİLİNİN AERODİNAMİK ÖZELLİKLERİ ÜZERİNDEKİ HÜCUM KENAR DEĞİŞİMİNİN ETKİSİ

Tolga Baran ÖZDEMİROĞLU

Yüksek Lisans, Havacılık ve Uzay Mühendisliği Anabilim Dalı

Danışman: Doç. Dr. Tahir DURHASAN

Temmuz 2025, 106 sayfa

Bu çalışmada, NACA 0012 kanat profilinin aerodinamik performansının, şekil değiştirebilen hücum kenarı yöntemi ile artırılması amaçlanmıştır. Araştırma, 120 mm veter uzunluğuna sahip bir kanat üzerinde, Reynolds sayısının, $Re = 10^5$, koşullarında gerçekleştirilmiştir. Yalın kanat profiline ek olarak, sapma açısı ve sapma başlangıç noktası (veter uzunluğunun yüzdesi olarak ifade edilmiştir) giriş parametresi olarak alınarak değiştirilmiş geometriler oluşturmak üzere bir MATLAB kodu geliştirilmiştir. Hesaplamalı analizde, sapma açıları 2° , 4° ve 6° olan ve sapma başlangıç noktaları veter uzunluğunun %10, %20 ve %30'unda bulunan hava profilleri değerlendirilmiştir. Tüm konfigürasyonlar ANSYS FLUENT yazılımı kullanılarak analiz edilmiş ve aerodinamik olarak sıfır sapmalı (0°) yalın durumla karşılaştırılmıştır. Performans değerlendirmesi, kaldırma katsayısının (C_L) maksimize edilmesi, sürüklenme katsayısının (C_D) minimize edilmesi ve sonuç olarak aerodinamik verimliliğin bir göstergesi olarak kaldırma/sürüklenme oranının (C_L/C_D) optimize edilmesi üzerine odaklanmıştır. Sonuçlar, 6° saptmaya sahip kanatların, stol (tutunma kaybı açısı) sonrası rejimde (örneğin $\alpha = 18^\circ$) yalın duruma kıyasla en yüksek aerodinamik verimliliği sağladığını ve laminar ayrılma balonunun azaldığını, çıkış kenarındaki girdapların ise zayıfladığını göstermektedir. Orta hücum açılarında ($\alpha = 10^\circ$), 4° sapmalı kanat profili en iyi performansı sergilemiş ve C_L/C_D oranında temel modele göre %73,61 iyileşme elde edilmiştir. Stol karakteristiklerinde sınırlı bir iyileşme gözlemlenmiş olsa da, tüm modifiye edilmiş geometriler yalın modele kıyasla daha düşük sürüklenme göstermiştir.

Anahtar Kelimeler: NACA 0012, Sürüklenme azaltma, Akış Kontrolü, şekil değiştiren hücum kenarı, sapma

ABSTRACT

THE EFFECT OF LEADING EDGE MORPHING ON THE AERODYNAMIC CHARACTERISTICS OF NACA 0012 AIRFOIL AT LOW REYNOLDS NUMBER

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July 2025, 106 pages

In this study, it is aimed to enhance the aerodynamic performance of the NACA 0012 airfoil with the morphing leading edge method. The investigation was conducted using an airfoil with a chord length of 120 mm under Reynolds number conditions of $Re = 10^5$. In addition to the base airfoil, a MATLAB code was developed to generate modified airfoil geometries by specifying the deflection angle and the deflection initiation point (expressed as a percentage of the chord length) as input parameters. The computational analysis considered deflection angles of 2° , 4° , and 6° , initiated at 10%, 20%, and 30% of the chord length. All configurations were analyzed using ANSYS FLUENT and compared aerodynamically with the base case (0° deflection). The performance evaluation focused on maximizing lift coefficient (C_L), minimizing drag coefficient (C_D), and optimizing the lift-to-drag ratio (C_L/C_D) as a measure of aerodynamic efficiency. The results demonstrate that airfoils with 6° deflection achieved the highest aerodynamic efficiency relative to the base model in the post-stall regime (e.g., $\alpha = 18^\circ$), with observed reductions in laminar separation bubbles (LSB) and trailing-edge vortex intensity. At moderate angles of attack ($\alpha = 10^\circ$), airfoils with 4° deflection exhibited optimal performance, yielding a 73.61% improvement in C_L/C_D compared to the base model. While stall characteristics revealed limited improvement, all modified geometries demonstrated reduced drag compared to the base model.

Keywords: NACA 0012, Drag reduction, Flow Control, Morphing Leading Edge, Deflection.

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LIST OF ABBREVIATIONS

TKE	: Turbulance Kinetic Energy
HAC	: High Aspect Ratio Configuration
LSB	: Laminar Speration Bubble
HSC	: High Swept Configuration
CFD	: Computational Fluid Dynamics
FSI	: Fluid Structure Interaction
SMA	: Shape Memory Alloys
LES	: Large Eddy Simulation
RSM	: Reynolds Stress Model
MCL	: Mean Camber Line
VSMTW	: Variable Span Morphing of Tapered Wings
CL	: Chord Line
OSPL	: Overall Sound Pressure Level
AR	: Aspect Ratio
HECS	: Hyper Elliptic Cambered Span
FEA	: Finite Element Analysis
MFC	: Macro Fiber Composites
DMLE	: Dynamic Morphing Leading Edge
FEM	: Finite Element Method
NiTi	: Nickel-Titanium

MAV	: Micro Aerial Vehicle
MTE	: Morphing Trailing Edge
UAS	: Unmanned Aerial Systems
CFRP	: Carbon Fibre Reinforced Plastic
VG	: Vortex Generator
SST	: Shear Stress Transport
DMLE	: Dynamic Morphing Leading Edge
CMW	: Camber Morphing Winglet
FGW	: Fixed Wing Geometry
RSM	: Reynolds Stress Model
SA	: Spalart Allmaras
LEV	: Leading Edge Vortex
UAV	: Unmanned Aerial Vehicle
DRS	: Double Rib Sheet
VSMW	: Variable Span Morphing Wing
RANS	: Reynolds Averaged Navier Stokes
PIV	: Particle Image Velocimetry

LIST OF SYMBOLS

ρ	: Fluid density (kg/m ³)
μ	: Dynamic viscosity (Pa·s or N·s/m ²)
D	: Characteristic dimension (wing chord, m)
ν	: μ/ρ = kinematic viscosity (m ² /s)
x/c	: Location on an airfoil with respect to chord percentage
A	: Reference Area
V_∞	: Freestream Velocity
ρ	: Air density
F_D	: Drag force
F_L	: Lift Force
P	: Static pressure
P_∞	: Free flow static pressure
θ	: Deflection Angle
k	: Turbulence kinetic energy
μ_t	: Turbulence viscosity
P_k	: Generative turbulence kinetic energy
β^*	: Model constant
σ_k	: Model constant
$\rho \vec{g}$	: Gravitational force
$\bar{\tau}$	: Stress tensor
C_L	: Lift Coefficient
C_D	: Drag Coefficient
C_P	: Pressure Coefficient
C_L/C_D	: Lift to drag Ratio
St	: Strouhal Number
Re	: Reynolds Number

1. INTRODUCTION

In recent years, morphing leading edge technologies have gained significant attention in aerodynamic research, especially for applications involving low Reynolds (Re) number flows. Flows characterized by Reynolds numbers less than approximately 5×10^5 are typically referred to as low Reynolds number flows and are widely used in micro aerial vehicles, unmanned systems, and small-scale aircraft. One of the most notable features of these flow regimes is the formation of laminar separation bubbles (LSBs), which occur when the boundary layer separates while still laminar, then reattaches after a short distance, often causing a localized adverse pressure gradient and flow instability. These bubbles can lead to significant aerodynamic performance degradation, including increased drag and decreased lift-to-drag ratios. Therefore, these undesirable flow conditions must be controlled and reduced to the desired state. Morphing leading edge designs offer a promising approach to mitigate such issues by dynamically altering the geometry to control separation behavior and improve flow attachment. Moreover, morphing wing can increase aerodynamic efficiency by performing systematic changes in wing geometries. The idea of morphing wings was put forward as a result of the excessive increase in global gas emissions, especially after the second half of the 20th century. Advancements in the intelligent material such as SMA (Shape Memory Alloy), MFC (Macro Fiber Composites) provides the design of morphing wings.

Since the wing has a dynamic structure in a fixed wing aircraft, there are control surfaces in certain parts of the wing surface. These control surfaces change the momentum of the flow by giving various disturbances to the region called the boundary layer just above the wing wall. Therefore, the desired flow conditions are achieved. There are mainly 5 types of control surfaces in airplanes. These are flap, slat and aileron, elevator and rudder, and there are also 3 types of axes of motion. These are roll, yaw and pitch. Figure 1.1 shows the schematic representation of the axes. The roll axis is the longitudinal axis of the aircraft and can make this movement thanks to the ailerons. The yaw axis is the lateral axis of the aircraft. The rudder mechanism is used to move on this axis. The pitch axis is the vertical axis of the aircraft. This axis is moved on this axis during gaining or losing altitude. The control surface of this axis is the elevators. Flaps are one of the most basic control surfaces in fixed wing airplanes. They are usually placed on the trailing edge and increase the carrying force of the wing during flight phases where high lift is required, such as takeoff and landing. They fulfill this task by

increasing the camber of the wing. On the contrary, they are also used to reduce altitude by reducing the lift. Since the lift force is increased in landing and take-off, it is possible for the aircraft to travel at a lower speed, but since the wing area and camber are increased, the aerodynamic friction force increases, which increases the fuel consumption of the aircraft. The slat mechanism is usually placed on the leading edge, allowing air flow just above this edge. In this way, as the air approaches the wing in the upstream region of the flow, the air molecules bond to the wing surface from the leading edge and flow separation is largely prevented.

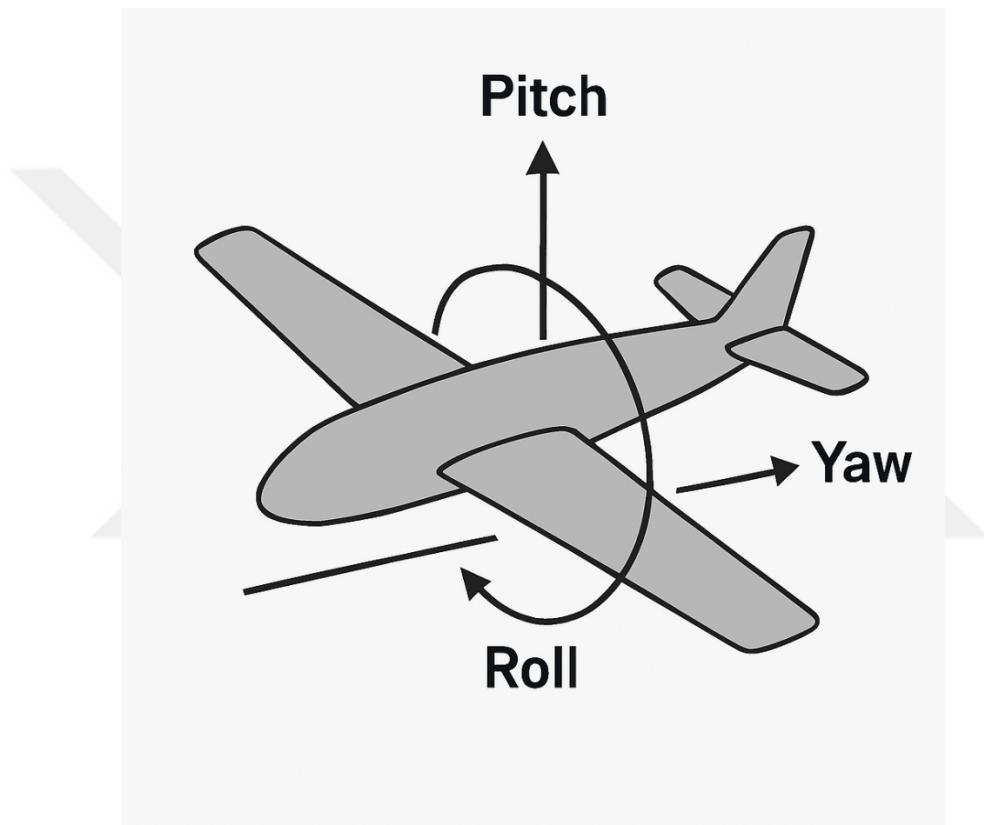


Figure 1.1. Motion axis in an aircraft.

At high angles of attack, the flow cannot attach to the wing surface and separates, resulting in reduction in lift. The slat mechanism increases the stall angle of attack, allowing the aircraft to have a safe flight characteristic at higher angles of attack. The slat mechanism ensures that the flow adheres to the wing at low angles of attack, takeoff and landing, but production costs are high due to the complex systems. In addition, power consumption is inevitable as they cause the wing to become heavier. Figures 1.2 and 1.3 show images of the flap, slat and aileron. Ailerons are control surfaces that are generally positioned close to the wing tip and allow roll

movement by creating moment on the roll axis. It is critical for the maneuverability of the aircraft. However, in a left or right roll, the wing in the opposite direction to the roll direction is subjected to more drag than the wing in the roll direction because the speed of the outer wing is higher than the inner wing. This results in an undesired yaw motion in the airplane. This is called adverse yaw.

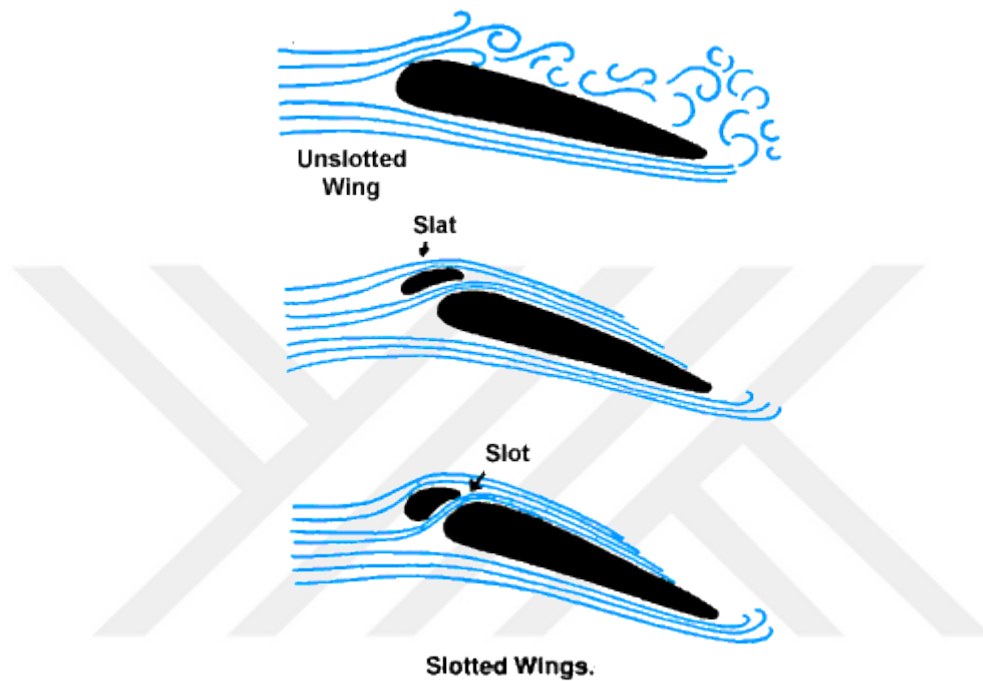


Figure 1.2. Slat mechanism in a wing [1]

Adverse yaw motion can be prevented by the rudder control surface. Figure 1.4 shows a visualization of adverse yaw motion. All these classical control surfaces have been used almost since the early days of aviation. However, when these surfaces are integrated into the wing, even when they are not used, they can disrupt the structure of the flow and adversely affect the aerodynamic performance of the aircraft. This leads to increased fuel emissions.



Figure 1.3. Illustrates aileron and flap control panels in an aircraft [2]

Morphing or deformable wings are one of the systems that have been proposed as a solution at this stage. Actively controllable morphing wings can activate the flow structure when desired and return to its original state when the task is completed. In addition, although systems such as slats make the wing quite heavy, morphing wings can be integrated into the wing in a much more compact and lightweight manner with advanced avionics circuits. In addition, the production costs of many morphing wings are less than systems such as slats.

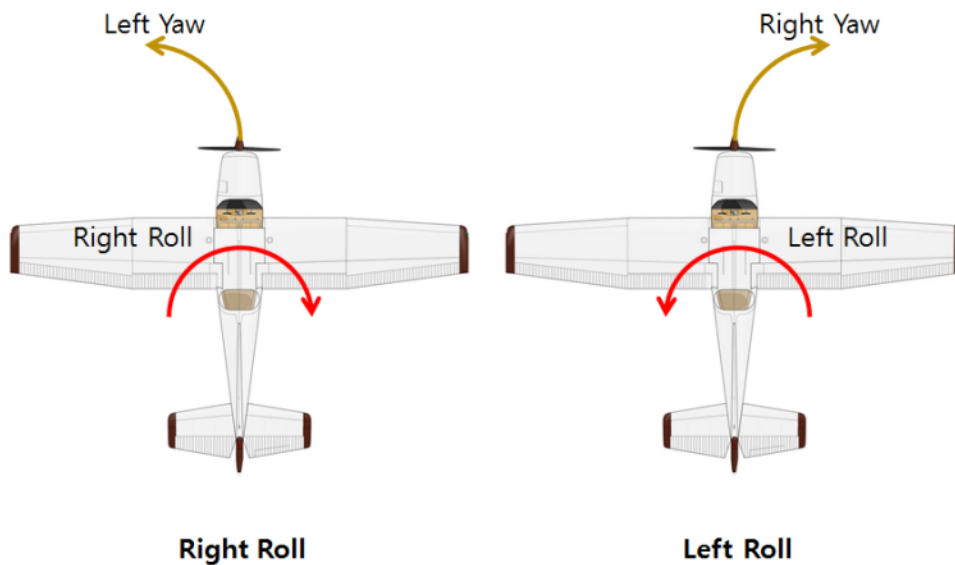


Figure 1.4. Adverse yaw whilst rolling right and left [3]

1.1. Morphing Wing

Advancements in aerospace engineering increasingly demand higher aerodynamic efficiency, improved maneuverability, and adaptive performance under varying flight conditions. One promising approach to meet these versatile goals is the implementation of morphing wing technologies. Morphing wings are aerodynamic surfaces capable of changing their shape, configuration, or structural properties in a controlled and continuous manner during flight. Unlike conventional fixed-geometry wings, morphing wing systems enable real-time adaptation to external aerodynamic and operational conditions, thereby enhancing overall aircraft performance across multiple flight regimes. The concept of wing morphing is fundamentally inspired by nature, particularly by birds and insects that dynamically alter their wing shapes to optimize lift, drag, and control forces during various phases of flight. Translating these biological principles into engineered systems has led to the exploration of various morphing mechanisms, including variable camber, span extension, twist adaptation, and leading/trailing edge shape transformation. Among these, morphing the leading edge has shown particular promise in delaying flow separation and reducing aerodynamic penalties in low Reynolds number environments, where conventional control surfaces may be less effective because of high flow instability. Structurally, morphing wings pose significant design and integration challenges. The system must exhibit a high degree of flexibility and actuation responsiveness while simultaneously maintaining sufficient stiffness to withstand aerodynamic loads. Materials such as smart composites, shape memory alloys (SMAs), and flexible skin structures have been extensively studied to reconcile this dual requirement. Moreover, the integration of morphing mechanisms often necessitates sophisticated control algorithms and sensor systems to monitor and optimize performance dynamically. From an aerodynamic perspective, morphing wings can be tailored to minimize drag, maximize lift-to-drag ratio, and adapt airfoil characteristics to specific mission profiles. This makes them particularly attractive for unmanned aerial vehicles (UAVs), micro aerial vehicles (MAVs), and other aircraft that operate under variable speed and load conditions, including those in low Reynolds number regimes. The strategic implementation of morphing wing concepts can significantly influence not only aerodynamic efficiency but also mission adaptability, stealth characteristics, and fuel economy. Given these advantages, a wide array of morphing wing strategies has been proposed in the literature, each offering distinct trade-offs in terms of complexity, performance, and feasibility. The first example of this technology can be considered twist morphing, which was

used by the Wright brothers in their airplane in 1903. Figure 1.5 shows a visualization of the technique used by the Wright brothers.

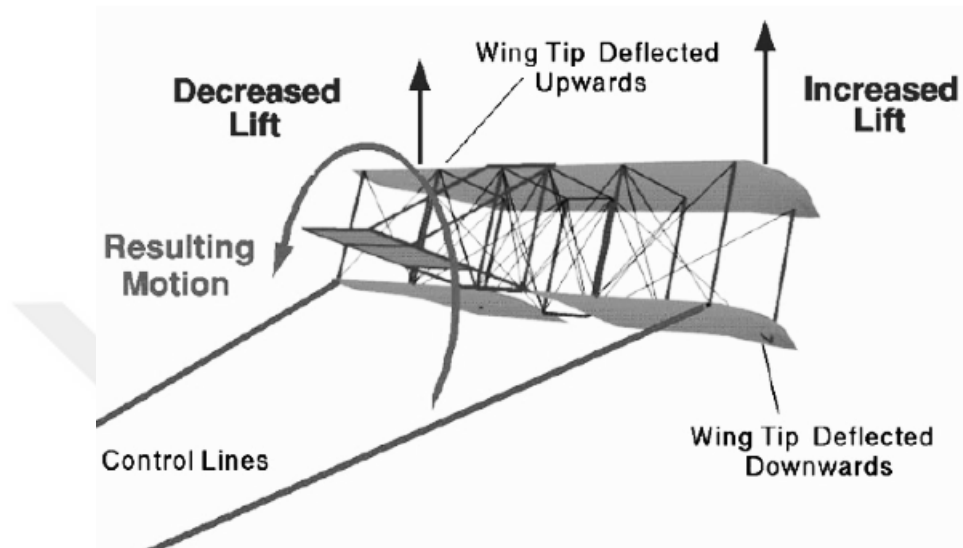


Figure 1.5.Wright Brothers using Twist Morphing in their wing[4]

1.2. Classification of the Morphing Wing

Morphing wing methods can basically be divided into 3 main sections according to the area where the morphing process is applied. Figure 1.6 shows the morphing wing shapes in various applications.

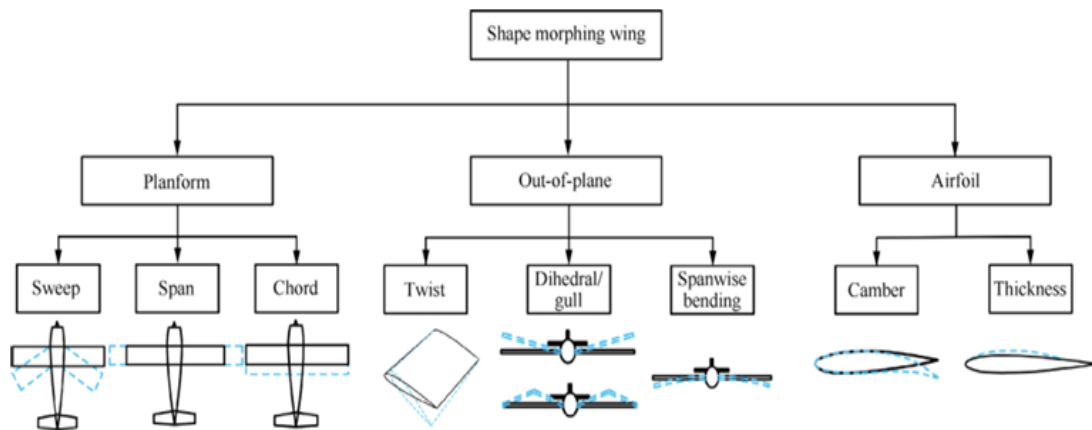


Figure 1.6. Morphing Types of Planform and Out of plane Morphing [5]

1.2.1. Planform Morphing

If the changes made on the wing take place on a certain plane, this is called planform morphing. These changes usually take place in two dimensions. There are 3 types of this method. These are span morphing, chord morphing and sweep angle morphing.

1.2.1.1. Span Morphing

Span Morphing was one of the first methods that engineers and researchers worked on in shape-changing wing technology. With this method, the length of the wingspan can be adjusted in different dimensions under different flight conditions. A mechanism, usually placed around the wing tip, increases the wingspan by activating the system during flight when a high wingspan is desired. The advantage here is that the system can increase the wing Aspect Ratio (AR) value. High AR reduces the aerodynamic drag of the vehicle by reducing wing tip vortices, which positively affects fuel economy. In addition, high AR increases the stability of the vehicle. This is a highly sought-after feature in aircraft where high stability is required, such as training aircraft. The first aircraft to use Span Morphing was the Mak-10. This aircraft could extend its wingspan from 13 meters to 21 meters. In addition, since the AR can be increased in this system, glide performance can be increased, but the disadvantage of the system is the increased weight. Figure 1.7 shows the geometric drawing of the Mak-123 aircraft with span morphing.

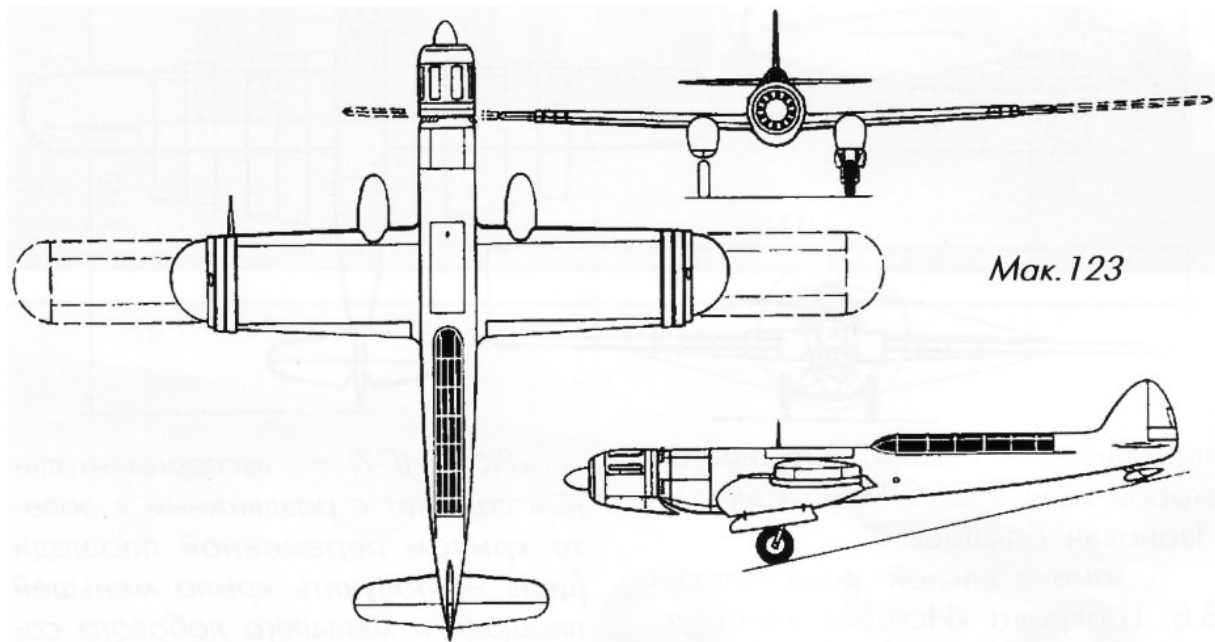


Figure 1.7. Aircraft Mak-123 having span morphing wings [6]

1.2.1.2. Chord Morphing

As in the Span morphing method, in the Chord Morphing method, the wing geometry can be played with and restored by changing the length of the wing chord instead of the wing span. In this method, as in a few other methods, the airfoil geometry is not changed, only the length of the chord is changed. With chord morphing, wing tip vortices can be reduced and aerodynamic drag and noise caused by this drag can be reduced, but system elements that are used continuously may wear out over time and their performance may decrease, so it may be necessary to use high-tech engineering materials such as SMA (Shape Memory Alloys). Figure 1.8 shows an illustration of the application of this method.

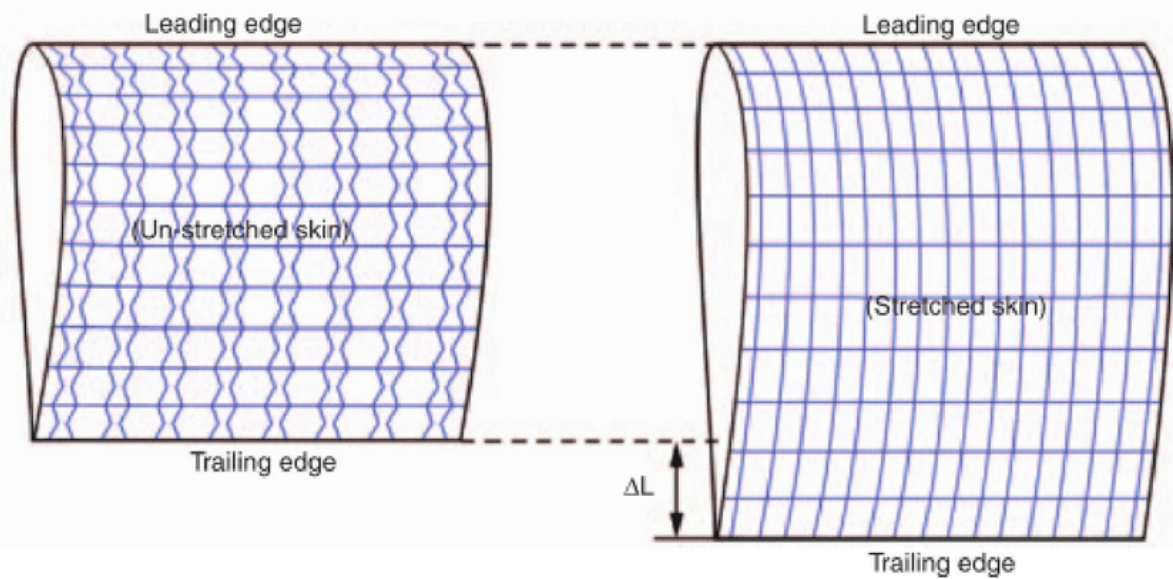


Figure 1.8. Schematic illustration of Chord Morphing[7]

1.2.1.3. Sweep Angle Morphing

Sweep angle has been one of the most widely used and effective wing deformation methods in fixed wing aircraft throughout history. Thanks to this method, especially fighter jets operating on aircraft carriers can be given a sweep angle to the aircraft due to the size of the wing areas and the lack of enough space on the ship to accommodate the aircraft, allowing the aircraft to move more easily and position them more comfortably on the ship deck. Sweep angle provides a healthier aerodynamic characteristic to the aircraft at high flight speeds. Figure 1.9 shows a North American F-86F with sweep angled wings. The sweep angle is usually given towards the rear, but in some cases, it can also be given towards the front. A LET L-13 glider with forward swept wing can be seen in Figure 1.10.



Figure 1.9. A North American F-86F with backward sweep modification[8]



Figure 1.10. LET L-13 glider with forward swept wing [9]

Airplanes with backward sweep angles generate mainly more lift, but airplanes with forward sweep angles better direct the airflow toward the airframe, so that the nose of the airplane can remain higher at higher angles of attack and even at stall angles. Airplanes with forward sweep angles are more difficult to operate than airplanes with backward sweep angles. In addition, when the sweep angle is given, the buckling moments at the wing root are reduced, especially in situations that expose the system to more gravitational forces, such as turning, and the structural integrity of the wing is ensured, and its strength is improved. The most important effect of the sweep angle on the aircraft and the reason why it is preferred is its ability to translate the critical Mach number on the wing. Since the average velocity of the airplane and local velocities at various points on the wing cannot be same, local Mach numbers are important to be considered for flow control perspective. The more the local Mach number can be delayed, the possibility of sound wall formation is prevented. A visualization of this is given in Figure 1.11 and a representative visualization of the variation of the critical Mach number with respect to the drag coefficient is given in Figure 1.12.

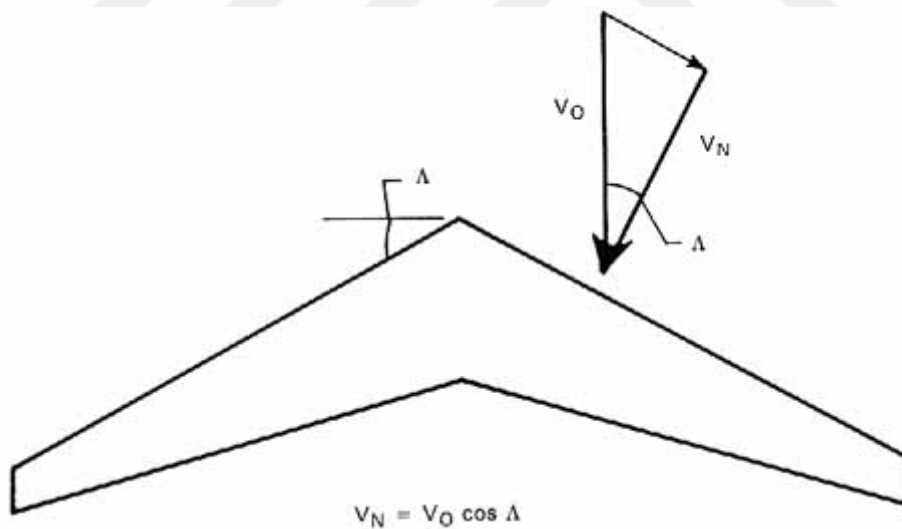


Figure 1.11. Decreasing upstream velocity by backward swept wing[10]

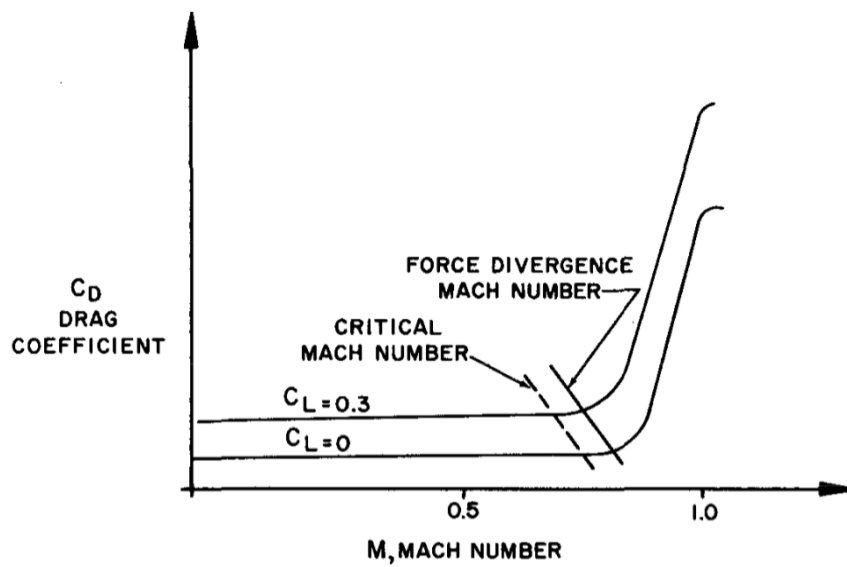


Figure 1.12. Graph of Critical Mach number versus C_D [11]

1.2.2. Out of Plane Morphing

In out of plane morphing method, the shape change occurs in 3 dimensions. The most iconic aircraft using this method is the XB-70, which can position the wing tips downwards. Figure 1.13 shows an image of this airplane.



Figure 1.13. Northon Xb-70 aircraft with downward deflected wing [12]

Out of Plane Morphing method is divided into 3 parts according to the way it is applied. These are Chordwise Bending, Spanwise Bending and Wing Twisting.

1.2.2.1. Chordwise Bending

The Chordwise Bending method is an old method used by the Wright Brothers in their airplanes in 1903. In this method, the mean camber line of the airfoil is changed in the entire part of the wing from the root part where the wing contacts the fuselage to the wing tip. It is possible to bend the wing upwards or downwards. With this change, different gaps on the wing surface are reduced as much as possible and the wing is made smoother. This allows the vehicle to use flight moments such as roll, pitch and yaw more efficiently and actively. It is also a very effective method of reducing wing weight as various control surfaces are eliminated from the wing, which has a positive effect on fuel economy. However, since it reduces flow separation and provides better adhesion of the flow to the wing surface, it both increases the lift coefficient and reduces the drag coefficient. Therefore, it increases the C_L/C_D ratio. A disadvantage of this method is that when the wing is exposed to aerodynamic stresses during flight, it becomes unstable. Therefore, advanced, reinforced and flexible materials need to be used to overcome potential adverse problems.

1.2.2.2. Spanwise Bending

In the spanwise bending method, the geometry is created by bending up or down from certain points of the wing. The bending is given by the joints located at these points. This method is not commonly used due to application difficulties. The materials used must be long-lasting and flexible. One of the most important features of this method is that it can give dihedral and anhedral to the wing by defining angles between the wing and the horizontal plane. When the wing is bent downwards or upwards, it is defined as dihedral or anhedral, respectively. In some cases, dihedral can make the aircraft more stable by increasing the rolling stability in lateral winds. Unlike dihedral, anhedral increases the maneuverability of the aircraft by reducing stability. This is a desired feature in aircraft such as fighter jets where high maneuverability is required. Figure 1.14 shows an image of a test aircraft spanwise bending in a real application, and Figure 1.15 shows two aircrafts with dihedral and anhedral wing types.



Figure 1.14. A NASA aircraft having spanwise bending wings [13]

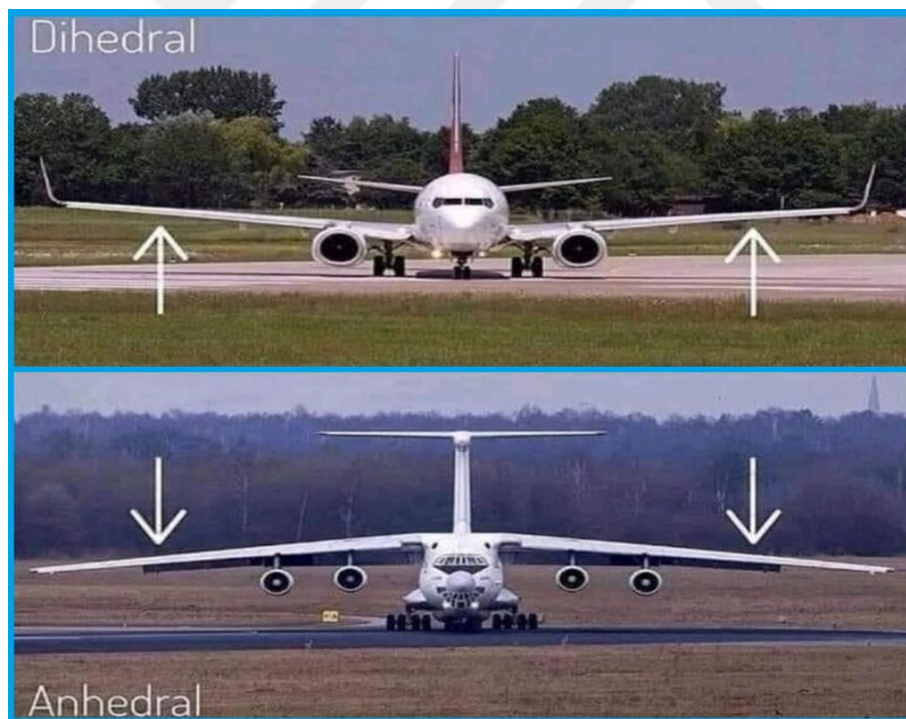


Figure 1.15. Dihedral and Anhedral wings on aircraft [14]

1.2.2.3. Wing Twisting

Wing twisting, unlike chordwise bending, is a method defined as changing the angle of incidence of the wing by keeping the shape of the airfoil constant over the distance from the wing root to the wing tip instead of changing the geometry of the airfoil. This method is generally applied to improve the stall performance of the aircraft. Figure 1.16 shows a wing twisting method. The stall condition is mainly expected to be at the wing tip. If the stall occurs at the wing root, the flow cannot be directed by the control surfaces and the stall condition may worsen and progress to the crash of the aircraft. Thanks to the wing twisting method, the stall condition of the flow can be moved to the wing tip by applying different incidence angles from the wing root to the wing tip. In this sense, wing twisting can be examined in two parts. The first is wash-out and the other is wash-in. If the incidence angle decreases from the wing root to the wing tip, it is called wash-out. On the contrary, if the incidence angle increases from the wing root to the wing tip, it is called wash-in. The Grumman X-29 due to its wings with a forward sweep angle, the wing root passes to stall first at high speeds, and this situation was tried to be eliminated by applying wash-in to the wing to prevent this. Figure 1.17 shows a visual of the wash-out application.

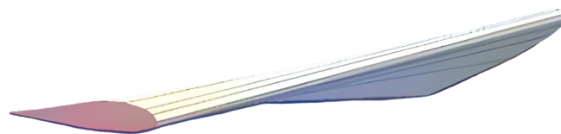


Figure 1.16. A twisted wing

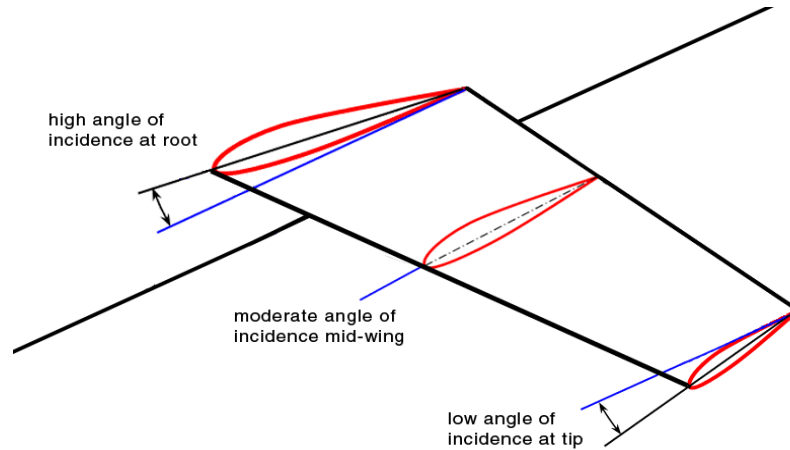


Figure 1.17. Schematic view of A Wash-out wing [15]

1.2.3. Airfoil Adjustment

In addition to planform morphing and out of plane morphing, there are also studies to improve the aerodynamic performance of the wing by directly modifying the airfoil. These are usually done by changing the camber or thickness of the airfoil. Camber can be defined as the distance between the Mean Camber Line (MCL) and the Chord Line (CL) on the airfoil. Variable camber wings are one of the most focused topics in aerodynamics today. With this method, optimum Aerodynamic efficiency (C_L/C_D) can be achieved without compromising aerodynamic performance under different flight conditions. It is possible to apply the camber on the airfoil at the leading edge, trailing edge or along the entire length of the airfoil, but each point of application has a different effect on wing aerodynamics. For example, the variable camber method applied at the leading edge prevents flow separation and increases the lift coefficient, while the variable camber method applied around the trailing edge reduces the wake area at the rear of the airfoil and contributes to the reduction of aerodynamic drag by reducing the formation of tip vortices. It is possible to apply the variable camber method on the airfoil with a wide variety of methods. The most prominent tools in this regard are advanced engineering materials such as piezoelectric activators and Shape Memory Alloys (SMAs). Piezoelectric activators are deformed as a result of external energy input and change the camber of the airfoil. SMAs, on the other hand, are very special metallurgical materials that can undergo shape changes when cold, but can return to their previous state when heated. It is possible to obtain

variable camber wings by coating the surface of the airfoil with these materials. Thickness is the distance between the upper and lower camber in the airfoil. Aerodynamic efficiency value can be increased by changing the thickness value. Figure 1.18 shows a visual of the important regions in an airfoil.

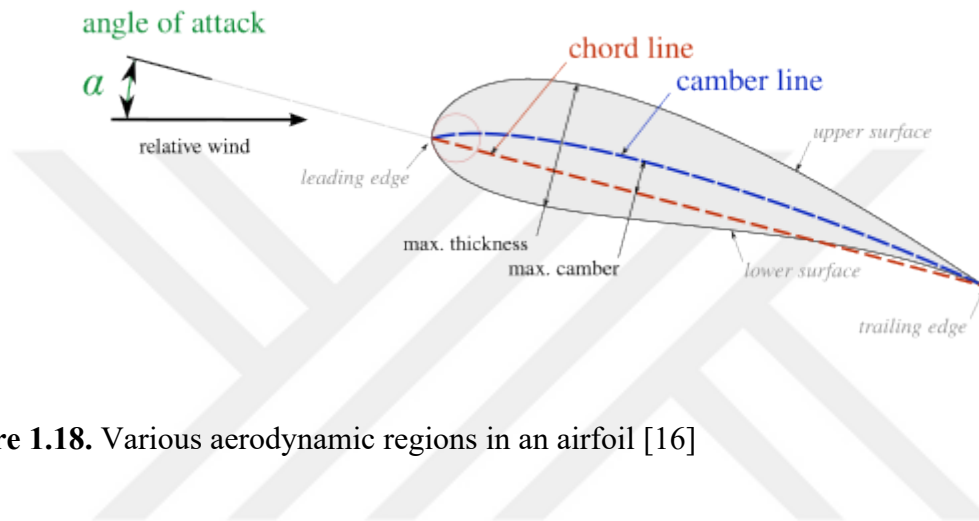


Figure 1.18. Various aerodynamic regions in an airfoil [16]

1.3. Flow Around an Airfoil at Low Reynolds Number

In flow around an airfoil, flow regimes can be classified depending on the Reynolds number. Reynolds number less than $Re=5 \times 10^5$ are defined as low Reynolds number flows, while flows above this value are defined as high Reynolds number flows. Low Reynolds number regimes ($Re < 10^5$), which are commonly encountered in micro air vehicles and small-scale UAVs, display distinctive characteristics compared to high-Reynolds-number flows. The dominant viscous forces and relatively weak inertial effects render the laminar boundary layer highly sensitive to even mild adverse pressure gradients, especially near the leading edge. As a result, the boundary layer may separate prematurely, giving rise to a laminar separation bubble (LSB) a phenomenon that plays a critical role in shaping the aerodynamic performance of airfoils in this regime [17,18]. An LSB forms when the laminar boundary layer separates due to a rise in pressure, followed by a transition to turbulence within the detached shear layer, and ultimately reattachment further downstream. This process creates a closed recirculation zone between the surface and the outer inviscid flow. The inner region of the bubble features reversed flow near

the wall, while the outer shear layer remains initially laminar but is prone to instability – most notably the Kelvin-Helmholtz type – which accelerates transition and facilitates reattachment [19]. The bubble typically consists of three zones: the separation point, the transition region, and the reattachment point. Its location and size are governed by several factors including Reynolds number, pressure gradient, and surface curvature[20]. Figure 1.19 illustrates the formation of laminar separation bubble.

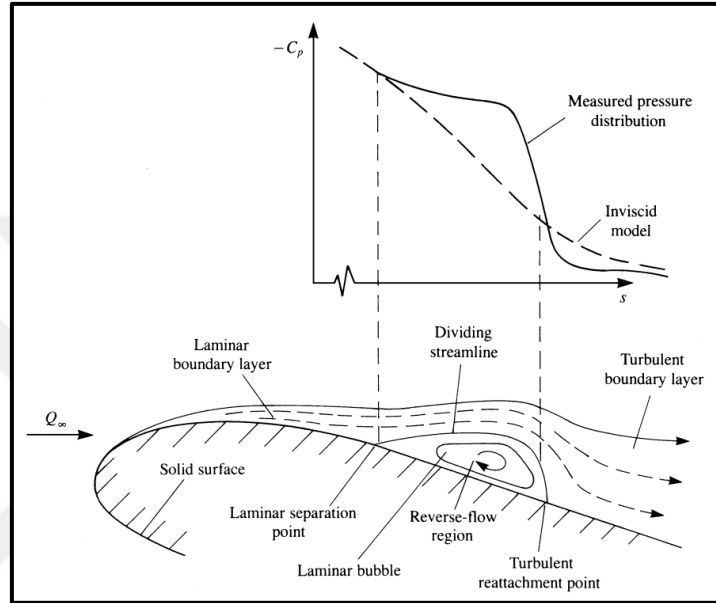


Figure 1.19. Formation of Laminar Separation Bubble with Representation of Pressure Coefficient [21]

At Reynolds numbers near 10^5 , LSBs often originate near the leading edge and terminate within the first half of the chord length, significantly modifying the pressure distribution and causing a pronounced suction peak. This, in turn, impacts both lift and drag, sometimes degrading aerodynamic efficiency, but also offering opportunities for flow control via geometrical or active strategies[18]. Understanding and predicting LSB dynamics is essential not only for performance optimization but also for the design of advanced flow control methods. Morphing leading edge technologies, for instance, offer promising potential to suppress or manipulate bubble formation by altering the boundary layer development in real time, especially under fluctuating operating conditions. The laminar separation point may occur as early as $x/c \approx 0.05$, with reattachment around $x/c \approx 0.3$, although these values are highly dependent on the airfoil geometry and freestream conditions[18]. The transition process inside the LSB can be

categorized as either short bubble or long bubble behavior. In short bubbles, transition and reattachment occur in close proximity, leading to relatively steady and compact recirculation zones. Long bubbles, in contrast, are associated with delayed transition and may exhibit strong unsteadiness or low-frequency oscillations—behaviors that can degrade airfoil performance or even trigger full flow separation under adverse conditions[22]. The formation of a laminar separation bubble (LSB) significantly alters the aerodynamic characteristics of an airfoil operating at low Reynolds numbers. One of the most prominent effects is the distortion of the pressure distribution, particularly on the suction side, where a pronounced suction peak may be observed near the leading edge due to flow deceleration and separation[18]. Although this suction contributes positively to the lift, the associated adverse pressure gradient downstream and the reattachment process can limit overall performance. The separated shear layer and the recirculation zone introduce a region of increased drag, primarily due to the enhanced pressure drag. Unlike turbulent separation, which tends to remain open and unsteady, the LSB remains relatively confined; however, its presence increases form drag while reducing skin-friction drag, resulting in a complex net effect on the overall drag coefficient [23]. Furthermore, the location and length of the LSB can influence the center of pressure and pitching moment. In cases where the bubble is unstable or oscillatory, the unsteady reattachment point may induce low-frequency oscillations in lift and moment coefficients commonly referred to as aeroacoustics instabilities or bubble bursting phenomena[22]. These fluctuations are especially critical in applications like micro air vehicles (MAVs), where control authority and stability are tightly coupled to small variations in aerodynamic forces. The Reynolds number dependency of the LSB also creates a narrow operational window for efficient airfoil performance. For instance, below a critical Reynolds number (typically around 70,000–120,000, depending on geometry), the LSB becomes longer, and transition delays can lead to full separation, resulting in dramatic lift loss and stall [17,18]. Modern airfoil design strategies, especially for low Reynolds number applications, often aim to control or delay the LSB, through passive or active techniques, such as surface roughness elements, boundary layer trips, or steady suction. The goal is to either eliminate the bubble entirely or shift transition upstream to minimize separation-induced drag [24]. The laminar separation bubble (LSB) has been a central topic in low Reynolds number aerodynamics for decades. Owing to its complex interplay between separation, transition, and reattachment, researchers have relied on both experimental and computational methods to understand its structure and behavior. Early experimental studies,

such as [17], employed oil flow visualization and hot-wire anemometry to identify the presence of LSBs on various airfoils. These works provided first insights into the bubble's location, length, and dependence on pressure gradients. Gaster's findings remain foundational, revealing that LSBs can undergo bursting if transition is delayed, causing sudden changes in aerodynamic forces. With the advancement of numerical methods, Direct Numerical Simulation (DNS) and Large Eddy Simulation (LES) became essential tools for LSB analysis. DNS studies by [19] demonstrated that the separated shear layer becomes unstable due to inflectional velocity profiles, which initiate transition via Kelvin–Helmholtz instabilities. Their simulations showed how short bubbles exhibit stable reattachment, whereas long bubbles lead to unsteady and sometimes chaotic behavior. Further DNS investigations by [25] emphasized the role of external disturbances, showing that even low-amplitude freestream fluctuations can significantly alter the size and frequency of LSB oscillations. These instabilities are important in practical applications, as they contribute to low-frequency unsteadiness, which can manifest as aerodynamic buffeting or tonal noise. On the experimental side, techniques such as Particle Image Velocimetry (PIV) and surface pressure measurements have been widely used to capture velocity fields and pressure distributions associated with LSBs. For example, [24] demonstrated the evolution of the bubble on a symmetric airfoil under varying Reynolds numbers and angles of attack. Their measurements clearly illustrated how transition onset affects reattachment length and lift behavior. In recent years, Reynolds-Averaged Navier–Stokes (RANS) simulations with transition-sensitive turbulence models (e.g., $k-\omega$ SST with $\gamma-Re_\theta$ transition) have made LSB predictions more accessible, especially in industrial CFD applications. Although RANS cannot resolve the full transitional dynamics as DNS or LES do, it offers a computationally efficient way to approximate bubble characteristics, particularly in parametric studies or design optimization. The downstream end of a laminar separation bubble is marked by the reattachment point, where the transitional shear layer impinges on the airfoil surface, restoring attached flow. This process is critically governed by the transition to turbulence within the shear layer, which increases the near-wall momentum and overcomes the adverse pressure gradient that initially caused separation [19]. The location of reattachment is not fixed; it fluctuates depending on Reynolds number, freestream turbulence, surface roughness, and airfoil geometry. In steady bubbles, reattachment is relatively stable, occurring at a predictable chordwise location. However, in many practical low-Reynolds number applications, the LSB exhibits quasi-periodic oscillations, leading to low-frequency fluctuations in the position of the

reattachment point [22]. These fluctuations can result in bubble bursting, a phenomenon where the reattachment process fails temporarily, allowing the separated region to expand dramatically. This behavior introduces significant unsteadiness in lift and moment coefficients, potentially compromising flight stability and control – especially critical in micro air vehicles (MAVs) or unmanned aerial systems (UAS) operating in low-turbulence conditions[18]. When the former dominates, the bubble remains short and stable. When the latter becomes more influential, due to either flow deceleration or insufficient perturbations, the bubble elongates and may become unstable or open, resembling a fully separated flow[25]. Control strategies targeting the reattachment dynamics aim to suppress or stabilize bubble oscillations. These include passive methods (e.g., boundary layer trips, vortex generators) and active methods (e.g., periodic suction/blowing). Successful reattachment control can delay stall, improve lift-to-drag ratio, and reduce sensitivity to freestream disturbances[24]. In low Reynolds number flows other than LSB, a structure called Von Karman Street may occur. The phenomenon known as Von Karman Street is one of the structures that can occur around the airfoil, which is known as Von Karman Street, which is named after Von Karman Street, who has worked in the field of fluid mechanics and high Reynolds number flows. Karman Street, which can generally be seen in bluff structures such as circular cylinders, can also be seen on wings under certain flight conditions. For this, low Reynolds number flow and relatively high angle of attack are required.

Figure 1.20 shows a visualization of the Karman street structure in the downstream region of a hydrofoil. From a physical standpoint, bubble dynamics are influenced by two competing processes. The first one is instability growth within the separated shear layer, which promotes transition and reattachment and the second one is adverse pressure gradient near the trailing edge of the bubble, which resists flow reattachment.

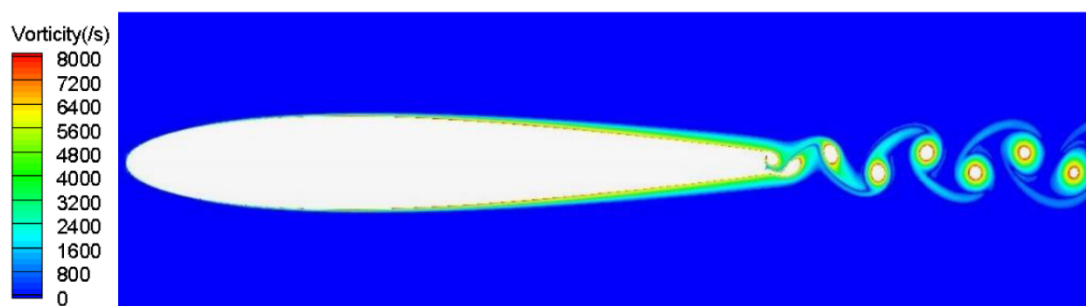


Figure 1.20. Von Karman Street formation on a hydrofoil [26]

1.4. Motivation of Study

Based on the aforementioned studies, it can be concluded that shape-changing or morphing wing models offer notable aerodynamic advantages, including extended flight endurance, improved lift generation during critical flight phases such as takeoff and landing, and a reduced reliance on traditional mechanical control surfaces. In addition, the seamless structure of morphing designs contributes to geometric simplicity and potential weight savings and these advantages are critical for the performance and efficiency of aerial vehicles. However, the studies on the aerodynamic characteristics of leading-edge morphing wings at low Reynolds number is limited. This gap is especially relevant for applications involving small-scale UAVs and micro-air vehicles operating in such flow regimes. In this sense, the present study aims to numerically investigate the aerodynamic effects of leading-edge morphing on the NACA 0012 airfoil at a low Reynolds number of 10^5 . Beyond performance metrics such as lift and drag, the study also seeks to visualize and analyze the flow structures around the airfoil to better understand how leading-edge morphing influences the velocity field, turbulent kinetic energy distribution, and streamline topologies.

2. LITARATURE REVIEW

The concept of morphing wings has gained increasing attention in recent decades as a response to the growing demand for energy-efficient, environmentally sustainable, and aerodynamically adaptable flight systems. Particularly after the 20th century, the aerospace industry began focusing on strategies to reduce fuel consumption and carbon emissions, while maintaining or enhancing performance across diverse flight conditions. Within this context, morphing wing technologies emerged as a promising solution to optimize aerodynamic characteristics dynamically, allowing aircraft to adjust their geometry in real time for improved lift-to-drag ratios, extended range, and enhanced maneuverability. In this method, the leading and trailing edges of airfoils can be changed. In previous studies, mostly research has been done on the trailing edge. Many technologies can be used in shape-changing wings. Among these, with the development of technology, SMA (Shape Memory) Alloy), MFC (Macro Fiber Composites). SMA is a type of alloy. As mentioned before, the morphing methodology is basically divided into 3 parts. These are planform morphing, Out of morphing and Airfoil morphing.

2.1. Planform Morphing

Planform morphing is the process of making changes to the areas seen when the aircraft is viewed from above. This can be done by changing the sweep angle, wingspan or chord length. Sweep morphing, which is a type of planform shape change, is one of the most used methodologies in wings. Although it has many effects, it is mostly used in transonic flights to reduce wave drag from sources exceeding the speed of sound and to increase the maneuverability of the aircraft. However, since it can reduce the wingspan, it provides an advantage to users in places where the usage area is very limited and important, such as aircraft carriers.

Yang et al.[27] designed and analyzed a deformable wing with variable sweep angle for UAV using the parallelogram mechanism. They designed the surface of the wing from kevlar and carbon fiber hybrid composite, an innovative material. The laminates were placed at 0° - 90° - 0° , that is, symmetric-cross-ply. Using the parallelogram method, structural features such as strain, stress and hardness coefficients on the composite material surface were analyzed numerically. According to the obtained model, the composite wing surface can change shape smoothly and

continuously when the sweep angle is between 20° and 60° . As the sweep angle increases, the wing area first increases but then decreases. When the sweep angle increases from 20° to 60° , the AR decreases by 118.1%. The wing root-wing tip ratio increases by 26.9%. Since the change in sweep angle is a dynamic event, it is important to control this event because with every change in the sweep angle, changes occur in the aerodynamic performance of the vehicle.

Marmier et al.[28] worked on inflatable, that is, pneumatic actuators for the sweep control of a small UAV. The wings of the test version are manufactured from the NACA 0012 airfoil, and activators that can be controlled by solenoid valves have been integrated into the vehicle. A test model was examined in the wind tunnel at a speed of 45 mph and different sweep angles. As the sweep angle increased, a decrease in the lift curve slope and AR value was observed. Since the area did not change much when the sweep angle changed at low angles of attack, no decrease in drag was observed. Moreover, it is found that the stall could be postponed with the sweep angle.

Dai et al.[29] studied the aerodynamic performance and range increase of the Waverider hypersonic spacecraft with a sweep wing configuration. They compared the models with each other by changing the sweep angle in different flight conditions. According to the results, at hypersonic speeds, variable sweep wing geometry increases aerodynamic efficiency in both flow regimes, although not as much as at supersonic speeds.

Si et al.[30] found that sweep does not reduce drag linearly and that the best aerodynamic efficiency is at a sweep angle of 45° .

Pflueger et al.[31] experimentally tested an aircraft with an elasto-flexible membrane in a wind tunnel. This vehicle can have different sweep angles and can be transformed into a high AR (Aspect Ratio) wing configuration. It was mentioned that asymmetric wing geometries transfer rolling moments better. PIV (Particle Image Velocimetry) was used to visualize the wing wake region. The high aspect ratio configuration (HAC) has an elliptical lift distribution while the model with high swept configuration (HSC) produced higher lift at the wing tip.

Pflüger et al.[32] numerically investigated that the aerodynamic effect of the elasto-flexible membrane wing at $Re=2.64 \times 10^5$. It was found that the deformation of the membrane has a direct effect on the lift force. Leading edge suppresses the vortex and makes it more stable thus shifting the stall angle.

Chen et al.[33] studied noise reduction on swept wings using droop nose. It was observed that the droop nose configuration reduced the overall sound pressure level (OSPL) of the mid-range frequency and the highest noise level by 15 decibels. With the droop nose configuration used, the highest lift to drag ratio increased by 2.7%. Moreover, the droop nose system used not only prevents the formation of vortex between the slat mechanism and the wing wall, but also prevents these two flows from mixing with each other. This allows the formation of a thinner boundary layer in the upper wing region and the flow becomes more laminar and wall pressure vibration decreases, thus reducing noise.

On the other hand, wingspan morphing, a type of Planform Morphing methodology, is a technique applied to achieve various aerodynamic gains on the wing by changing the wingspan of the vehicle. In this field, Bae et al.[34] conducted an aerodynamic and aeroelastic investigation of the Variable Span Morphing Wing (VSMW) of a cruise missile vehicle. In the light of the numerical findings obtained, as the wingspan increased, the lift distribution and induced drag increased to a certain extent across the wing length, but the bending moments at the wing root and the range of the missile also increased. However, it was stated that increasing the span contributed to the aerodynamic development of the missile, but weakened the missile aeroelastically due to bending stresses.

Guo et al.[35] designed a variable wingspan wing using Sarrus links and performed CFD and FEM analyses and found that although span morphing increases lift and drag by 50%, it has little effect on the lift and drag coefficient.

Bardera et al.[36] conducted a study on the numerical analysis of grids placed on wingtips for UAVs. They placed grids of different lengths on the wingtip and examined the effect of these attachments on aerodynamic and roll performance. This study, inspired by the wingtip feathers of birds, was found to reduce induced drag in most flight conditions except at very high angles of attack and negative angles. It was also found to increase roll capability except at very low angles. When the angle of attack was zero, the roll effect was very small.

Elelwi et al.[37] used Variable Span Morphing of Tapered Wings (VSMTW) wing technology and made a study in this field. They made a numerical study using different sweep angles at different altitudes and using the $k-\omega$ turbulence model. According to the results, an aerodynamic

performance increase of 32.39% was observed in the cruise phase. In addition, a 43.94% increase in vehicle range was achieved, which was achieved by significantly reducing drag. It was also stated that the roll rate value of the wings with varying wingspan decreased with increasing speed.

Bishay et al.[38] has studied the core part of a wing that can change the wingspan for use in unmanned aerial vehicles (UAV). The rib sections in this part are made of a rigid material. The strip parts are made of a flexible material. The spars, another part in the core part, are designed from carbon fiber material. The last part is the linear activator that performs the task of changing the wingspan. Thanks to this design, they were able to produce a wing that can increase the wingspan by 50%. The designed wing core was analyzed computationally in the SOLIDWORKS simulation program. As a result, it was seen that the linear actuator had sufficient force to operate the system. It was emphasized that the wings that can change the wingspan increase the lift force with extension and the fuel efficiency increases due to the increase in AR. It was mentioned that the maneuverability of the vehicle increased with the shortening of the wing length. As a result of changing the shape of each wing with different extension amounts, it was mentioned that it had a positive effect on the rolling ability of the UAV.

Ajaj et al.[39] conducted a conceptional study on wingspan shape change in order to provide the rolling performance to the vehicle by simply changing the wingspan during the 4 flight phases determined instead of classical aileron control surfaces. High-wingspan aircraft have high AR value, range and fuel efficiency, but low-wingspan aircraft have high maneuverability and high cruise speeds. As the wingspan increases, AR increases and the lift force distribution of the wing is low for a constant lift value, so the induced drag resulting from lift decreases, which increases the range and fuel efficiency of the vehicle, but with the increasing wingspan, the bending moments at the wing roots also increase and as a result, it was seen that the rolling moment created by the aileron surfaces could be created by applying asymmetric wingspan shape change. Although the total drag coefficient is lower than the aileron system, the bending moment at the wing roots is 60% higher as a result of asymmetrical, that is, extending one wing and keeping the other wing fixed.

Chord length change, which is another method of changing planform shape, is a method performed by changing the wing chord length according to flight conditions. In this method, no

change is made in the shape of the wing profile, only the chord length is shortened or lengthened. Due to the flexibility of the honeycomb structure, structural integrity could be achieved more easily.

Asins et al.[40] designed a morphing leading edge that can increase the chord length and wing camber in different flight conditions. This study can be included both in the planform morphing category due to the change in the chord length and in the airfoil adjustments category due to the change in the shape of the airfoil. CFD (Computational Fluid Dynamics) studies were carried out. The flow structures of wings and airfoils affect the acoustic layout of the structure. The droop nose system, which is an idea close to the shape-changing wing concept, is the process of leaving the area outside the leading edge of the airfoil fixed and lowering the leading edge at a certain angle. Apart from the leading edge, shape changes can also be made on the trailing edge.

2.2. Out-of-plane Morphing

Yu et al.[41] examined the bending and decoupled effects along the wingspan for flexible and thin wings. In this study, they examined and compared the decoupled effects of 4 different wing models using LES (Large Eddy Simulation) and FSI (Fluid Structure Interaction) methods. They used the FSI method to obtain a flexible wing to increase vehicle endurance. A new model was created by giving vertical deformations, that is, camber, along the wingspan of the model obtained with the FSI method, and it was analyzed in two wing thicknesses. It was observed that reducing the wing thickness by 40% caused a 15% decrease in drag. The model optimized according to the LES method contains upward dihedral and this model had high lift, high drag and high aerodynamic efficiency values compared to the other 3 models, but although this model had a higher lift coefficient than other models at all angles of attack, It had the highest drag at angles of attack up to 5° and the lowest drag at angles of attack of 10°.

Eguea et al.[42] analyzed the baseline model winglet structure and the optimized winglet structure with camber modification. According to the results, the optimized model brings the tip vortex closer to the trailing edge, but shock waves and flow separation did not affect the vehicle performance much, and this structure was found to reduce the vehicle's pressure drag and induced drag.

Eguea et al.[43] performed genetic optimization of the variable geometry winglet structure in order to increase the fuel efficiency of the business jet. For this purpose, the aerodynamic model was defined and the block diagram of the algorithm required for optimization was created. According to the results obtained, Fixed Geometry Winglet (FGW) had 6% less range compared to Camber Morphing Winglet (CMW), but it was able to carry 8 more passengers in the longest mission.

Zhang et al.[44] studied the aerodynamic effect of flapping trailing edge, one of the dynamic morphing methods, and obtained experimental and numerical results. They tested the morphing zone close to the trailing edge by flapping at frequencies between 10 and 90 Hz. The maximum deflection distance is 0.55 mm. Flapping at a frequency of 40 Hz increased C_L by about 2% and C_D by 1% and this method was found to be effective in improving aerodynamic efficiency.

Panagiotou et al.[45] studied the effect of cant angle in the winglet of a fixed wing UAV on vehicle aerodynamics. For this purpose, 7 different cant angle values were determined and the performances at these angles under different flight conditions were analyzed. The study was carried out in three steps. In step 1, a general analysis was carried out by including performance data and a range increase of 4.84% was achieved when the cant angle was 17.6. In step 2, stability conditions were taken into account and a range increase of 1.79% was achieved when the cant angle was 46.5°. In step 3, trimming was included and a range increase of 2.7% was achieved when the cant angle was 90°.

Manzo et al.[46] worked on the design and analysis of Hyper Elliptical Cambered Span (HECS) wing configuration to be used in aircraft. In the meantime, he examined many structural areas from the loads falling on the system to the wing loading. Analysis data were obtained for buckling caused by pressures, external and internal loads by performing FEA (Finite Element Analysis) on wings that can change shape thanks to SMA. The wing loading value of the designed wing was compared with the wings of other advanced commercial aircraft. It was revealed that the lift to drag ratio can be changed by at least 7% thanks to this wing structure produced. As a result, it was mentioned that it can compete with classical elliptical wings even in the test phase. However, it was emphasized that light activators should be used as much as possible due to the increase in wing weight due to system complexity and extra system equipment.

Chen et al.[47] used Variable Stiffness Spar (VSS) approach to increase the roll capability of the vehicle and analyzed at different speed values at altitudes up to 35×10^3 ft. Thanks to this approach, the stiffness of the spars can be changed and the wing can make different twist movements along the spanwise. The wings of a fighter jet were used as a baseline model and the roll capability and aeroelastic forces of the wings with this system were compared with the wings of the baseline model. The wing using the improved VSS approach had a 126% higher roll rate value than the baseline wing.

Nam et al.[48] similar to the previous study, selected the wings of the F16 fighter jet as the baseline model and a wing using spars made of SMA was proposed to increase the roll capability. ASTROS software was used to study the response of SMA actuators to roll motion, aeroelastic effects and roll efficiency at different speeds. It was mentioned that this technology provides torsional stiffness to the wing and has a higher roll performance than the baseline model. Especially in transonic flights, the roll performance of the baseline model was significantly reduced and the wing using this technology offered 61% higher roll performance than the baseline model.

Cooper et al.[49] aimed to change and improve the wing bending and torsional stiffness value with 2 approaches, either by rotating the wing spars or thanks to the movable structure. In addition, it is aimed to reduce the drag thanks to this modification. According to the results obtained after FEM analyzes and wind tunnel tests, it was revealed that drag can be reduced without sacrificing lift.

Cadogan et al.[50] used the inflatable wing method the inflatable wing method in order to improve the aerodynamic performance of the UAV but to flatten the peaks in certain regions of this geometry, a piezoelectric activator was used due to its short response time and sharp machining capability. These activators provide twist motion around the trailing edge. According to the results obtained after the prototype tests, when the trailing edge was deflected, the net lift force increased from 96 lb to 108 lb. It was also graphically demonstrated that this methodology increased the lift per unit span values.

Wang et al.[51] studied the energy efficiency and stability of the Albatros type deformed wings using anhedral. This anhedral drag coefficient at the wing tips was reduced by 7.75% and the maximum downwash intensity by about 30%. In addition, since this process increases the moment arm in longitudinal stability and decreases the moment arm in lateral stability, an

increase in these stabilities has been observed. Since it reduces the downwash by about 30% after the wrist point of the wing, the induced drag at the wing tip is reduced and therefore the vehicle endurance is increased by about 13%.

2.3. Airfoil Adjustment

Secanell et al.[52] designed the shape-changing airfoil using aerodynamic shape optimization. During this study, Spalart Allmaras turbulence model and quadratic programming algorithm were used to determine the optimum wing profile of the light unmanned aerial vehicle during all kinds of flight conditions. Using different airfoils for each flight situation increases the aerodynamic performance. For example, 1.6% camber value may be sufficient in wing profiles designed for cruise flight, while camber values of other wing profiles are used. It can go up to 7.9%. After these operations were carried out, the results were examined in detail in order to better understand the most efficient wing profile. According to the results obtained, it was concluded that the thin-thickness wing profile could be used first. In addition, it was emphasized that the leading edge thickness of the wing profile and the mechanism controlling the camber of the wing profile would provide almost sufficient performance in all flight conditions, and it was said that the use of the appropriate wing profile in various flight conditions would reduce energy consumption and consumption.

Namgoong et al.[53] made aerodynamic optimization between deformable airfoils and the activation energy required to change the shape of these airfoils. Multi-objective optimization was used to optimize the lowest activation energy and the least aerodynamic drag. In the light of the results obtained, the multi-objective optimization method improved energy efficiency and performance in various flow forms such as subsonic, transonic and supersonic. It has been found to be a suitable method in terms of low aerodynamic drag.

Bilgen et al.[54] conducted a study by integrating Macro Fiber Composite (MFC) into the wing profile for a ducted fan aircraft. The wing profile was examined experimentally in a wind tunnel with a flow speed of 15m/s. It was observed that MFCs showed a better performance in the field of aerodynamic control compared to classical control surface technologies. It was observed that as a result of the voltage applied to the MFC activators, the average lift coefficient could increase by 1.46.

Sodja et al.[55] discussed the morphing leading edge in a more structural sense in order to find the induced strain magnitude on the airfoil surface with the morphing leading edge. It was noted that deformable wings can reduce global carbon dioxide emissions by 50%, nitrogen oxide derivatives by 80% and noise by 50%.

Jawahar et al.[56] studied experimentally and computationally on variable geometry flaps on the NACA 0012 airfoil at the critical Reynolds number of $Re = 2.6 \times 10^6$. It was concluded that the deformable flaps on the trailing edge of the airfoil increase the aerodynamic performance as their camber increases. It increases the lift coefficient compared to standard flaps, but the C_L/C_D ratio is very low. However, it was observed that the flow separation near the trailing edge was delayed at high angles of attack.

Jones et al.[57] experimentally examined the flow separation of the NACA 4415 airfoil at low Reynolds number flow of $Re = 5 \times 10^4$. For this, macro fiber composite (MFC) was placed at the bottom of the suction part of the airfoil and by sending a voltage signal to these smart materials, the airfoil was moved up or down. It is observed that the high momentum created reduced separation and drag in the trailing edge and wake region and increases the lift force. In this way, the C_L/C_D ratio increases.

Abdessemed et al.[58] conducted a steady and time-dependent CFD analysis of the NACA 0012 airfoil with deformable trailing edges using the dynamic mesh method at $Re = 7.5 \times 10^5$. They made a comparison of deformable trailing edge flaps and flaps clamped to the trailing edge. At the highest deflection value (5% of the chord), the deformed wing profile created a higher lift force than the nullified flap, but the drag coefficient increased by a certain amount. The aerodynamic efficiency value (C_L/C_D) increased by 6.5%.

Achleitner et al.[59] designed a winged aircraft that can morph and the total performance of the aircraft's deformable wing was compared with a sailboat. The results showed a very significant increase in the glide ratio (C_L/C_D) of 8.5%, but it was emphasized that this performance increase could be achieved by very well activating the connections between the wing profile and the fuselage.

Kan et al. [60] numerically investigated the aerodynamic behavior of a deforming leading edge on a NACA 0012 airfoil using dynamic mesh techniques at a Reynolds number of 6×10^6 and angles of attack ranging from 0° to 17° . The study compared steady and unsteady aerodynamic

responses of the deformed and classical airfoils, showing that downward deflection of the leading edge increased the stall angle, lift coefficient, and C_L/C_D ratio prior to stall. While the aerodynamic impact of deflection was minimal at low angles of attack in steady flow, unsteady results revealed that transient lift response varied significantly with deflection direction and frequency. Overall, the morphing leading edge was found to effectively modify stall behavior and dynamic lift characteristics.

Suzuki et al.[61] aimed to reduce the drag of NACA 63a012 airfoil at $Re=3 \times 10^6$ flow conditions by using the leading edge morphing method. It was found that the lift to drag ratio increased significantly compared to the baseline model when the lift coefficient was greater than 0.3. When $C_L = 0.5$, the lift to drag ratio increased by 26%.

Communier et al.[62] conducted an experimental study on the deformation of the trailing edge of the airfoil and the aerodynamic control capabilities of the classical aileron mechanism and MTE (Morphing trailing edge) were compared. It is found that the lift coefficient increases as the angle of attack increases. However, drag analysis showed that MTE produces less drag coefficient for the same lift coefficient compared to the aileron mechanism. This shows that the L/D ratio is higher in MTE, which means it makes a positive contribution to fuel economy. In addition, with the use of this activator system, the total weight of the system is kept at reasonable values, a relatively complex system such as aileron is disabled, and fuel economy is improved.

Bashir et al.[63] carried out the design and optimization of a UAS-S45 airfoil with a droop nose leading edge within the scope of a certain project. They showed that droop nose leading edges increase the aerodynamic performance. The C_D of the optimized morphing airfoil containing the droop nose was 12.18% less than the normal version. Aerodynamic efficiency (C_L/C_D) was 15.22% higher in the optimized airfoil. The aerodynamic endurance function increased by 10%.

Bashir et al.[64] made a dynamic stall analysis of the airfoil that can change shape dynamically. UAS-S45 was chosen as the airfoil used. They numerically analyzed the unsteady flow using SST (Shear Stress transport) turbulence. To change shape of the airfoil, it was deformed from 15%, 20% and 25% of the chord distance, but various deflection values were applied at different angles of attack. Computational studies have shown that the airfoil leading edge curvature geometry prevents the formation of undesirable pressure gradients along the flow and reduces

the vortex intensity as a result of flow separation. The deformable airfoil delayed the stall angle by 14.26% and increased the highest obtainable lift coefficient from 2.49 to 3.04.

Camacho et al.[65] conducted a study to reduce the fuel consumption of the vehicle and increase efficiency by performing real-life optimization of the movable leading edge. An experimental study was conducted at Reynolds value of 2×10^5 . It was observed that changing the shape of the leading edge while in the stall initiation zone had a negligible effect on the lift and drag values. However, it still had a significant effect on the pitching moment. The leading edge, which can move when the wing is stalled, reveals a high difference of 45% in the lift coefficient, and the critical angle of attack also increases by a significant difference. This increases the aerodynamic ability of the wing, allowing it to provide sufficient lift even at higher angles of attack.

Bashir et al.[66] carried out the numerical analysis of the NACA 0012 airfoil with its leading edge and trailing edge, which can change shape in the transient flow corresponding to 2.5×10^6 Reynolds number. The results showed that the deformation of the trailing edge is very effective in increasing the lift coefficient, and the downward deformation of the leading edge is very effective in improving the lift coefficient after stalling. However, the shape change with low frequencies allows for higher lift coefficients and more stable flow.

Jia et al.[67], on the other hand, examined the formation of the Leading Edge Vortex (LEV) structure if bending was given along the wing span of the airfoil, which is in the form of a flat plate, which is easier to apply, instead of the classical airfoil geometries. The airfoil, accelerated from rest to $Re = 2400$, was tested numerically and experimentally by giving a 30° angle of attack. Here, two important effects of the spanwise bending method on the plate were observed. The first one reduces the formation of LEV by changing the vortex progression in the bent region of the plate. The second is that the plate creates a relative velocity field in the shear layer. Therefore, it prevents the formation of LEV to a certain extent.

Zhao et al.[68] made a numerical analysis using the DRS (Double Rib Sheet) structure, which will increase the efficiency when used in UAVs (Unmanned Aerial Vehicles), and produced the UAV prototype and tested it in comparison with the fixed-wing prototype. It is observed that the cruise flight time of the prototype with variable camber was obtained to be 4 minutes and 30 seconds longer than the flight time of the prototype with a fixed curvature angle of 3° . In the

light of all these data, it was understood that the variable camber wing increased the flight efficiency by 14.1%.

In some of the studies, investigations were made based on both wingspan and camber changes. Vale et al.[69] designed two interlocking wing structures, that is, a wing with a telescopic layout, and performed a number of CFD analyses, comparing a modified fixed wing and a telescopic wing structurally and aerodynamically. They observed that variable camber in the telescopic wing form positively affects the lift force. Moreover, changing the wingspan provides reducing the induced drag by increasing the AR (Aspect Ratio) value. Besides, the maximum climb rate decreases and drag increases by 4.15%. Stall speed decreased by 36.24% and endurance increased by 59.74%.

Davari et al.[70] studied the aerodynamic performance under flow conditions varying from Re 5×10^4 to 5×10^5 by changing the geometry of three different airfoils for the blades of small wind turbines. The results showed that the lift-drag ratio of some airfoils increased by 15-20%.

Wolff et al.[71] investigated the effect of the trailing edge, which can morphable, on the turbine performance for wind turbine applications. It has been shown that the lift change caused by morphing depends on factors such as the shape of the trailing edge, its camber, and its deflection angle. At the end of the study, there was a 15-20% increase in the lift coefficient.

Franco et al.[72] worked on a turbine with a deformable trailing edge in order to improve the performance of the wind turbine. When comparing turbines with fixed blades and deformable blades, it was found that the load decreased by 30% and the pressure coefficient increased by 15% in the turbine with deformable blades.

Abate et al.[73] conducted a study of the flow structure by adding tubercle structures found on the ends of whales' fins to the leading edge of the Cessna 172 Skyhawk aircraft. They found that these structures improved the lift to drag ratio at high angles of attack such as 15.5° .

Lyu et al.[74] the aerodynamic shape optimization of the morphing trailing edge wing was performed by considering different flight speeds and different altitudes, and also analyzed the aerodynamic data at different flight stages. According to the results, the fully morphing wing can reduce the drag coefficient by 6.2% at an altitude of 25×10^3 ft compared to the baseline model, while the morphing trailing edge wing can reduce the drag coefficient by 4.9% at the

same altitude. Finally, in general, fuel consumption can be reduced by about 1% at different flight trajectories.

Kaul et al.[75] carried out a numerical analysis of variable camber trailing edge flaps using two turbulence models and the results were evaluated. This model generated higher lift to drag ratio and lift values than the baseline model, but the stall angle of the baseline model was 2° higher. Variable camber flaps improved the pressure distribution and increased the pressure coefficient.

Bain et al.[76] designed a variable droop nose leading edge and tested it computationally on the helicopter blade. According to the results obtained from the modified blades designed with the code, the rotor efficiency increased by 2.9% and the 4/rev component, which causes vibration in the blades, was reduced by 8%.

Chandrasekhara et al.[77] studied the prevention of dynamic stall with the variable droop leading edge method provided that the VR-12 airfoil is within 25% of the chord length. They found that reduce the aerodynamic drag up to 75% compared to the classical model thanks to dynamically morphing leading-edge shape with different frequencies. It also noted that shape-changing or morphing wing models have many aerodynamic advantages such as enhanced flight time, sufficient lift force during takeoff and landing, and simplicity compared to classical control surfaces. Kumar et al.[78] demonstrated that drag reduction can be achieved up to 15.17% by camber changing of airfoil.

Kan et al.[79] aimed to improve aerodynamic performance of NACA 0012 especially at stall angles by applying periodic deflections to the trailing edge of the airfoil. According to the numerical results, at low angles of attack, the lift coefficient produced by dynamic deflection is lower than the static case and the lift coefficient decreases as the deflection rate increases. At high angles of attack, the opposite is the case.

Guo et al.[80] used a simple airplane model in order to determine the contribution of flexible wing geometry to vehicle aerodynamics in which the wing geometry is modified by the membrane. The geometry was also subjected to wind tunnel tests and the stall angle was shifted by about 5° and the lift coefficient increase was 90%.

Jo et al.[81] similarly, demonstrated improved vehicle aerodynamics by changing the wing camber. By changing the camber between 4 and 8%, a 17% flight range improvement is

observed when the baseline airfoil behaves like the NACA 6410 and a 60% flight range improvement is observed when the base airfoil behaves like the NACA 8410 airfoil.

Bashir et al.[82] analyzed the flow structures of an airfoil in stall state with the Dynamic Morphing Leading Edge (DMLE) method. They found that the flow separation was delayed by reducing the backpressure gradient and the stall angle was postponed by 16.58%.

Kumar et al.[83] investigated the effect of MTE (Morphing Trailing Edge) method for Micro Aerial Vehicle (MAV) at different angles of attack and Reynolds numbers. They performed experimental and numerical study and showed that lift coefficient increases with increasing the MTE deflection angle.

Wang et al.[84] investigated the effect of thickness and camber variation on flow stability in supercritical airfoils. It was emphasized that flow instability occurs in these airfoils with the change of camber angle and thickness value and unsteady effects increase as the deformation magnitude and frequency value increase. However, it was observed that the camber change was more effective in the formation of the unsteady state than the thickness change.

Murayama et al.[85] worked on variable flap technology for use in low Reynolds number flows and especially in small Unmanned Aerial Vehicle (UAV). These flaps are placed on the Naca 0012 symmetrical wing and covered with 30, 50 and 80 micrometer thick feathers using a bio-inspired, biomimetic approach. They proved that this technology can be used as a passive flow control method for flow around the airfoil.

3. MATERIAL AND METHOD

In the study, a 2D numerical study was conducted using ANSYS Fluent package program in order to investigate the effect of the leading edge morphing on the aerodynamic characteristics of airfoil. NACA 0012 airfoil section was used as reference case and leading edge morphing was applied to the NACA 0012 airfoil. All simulations were performed at Reynolds number of $Re=1 \times 10^5$ which is based on the chord length of the airfoil. The aerodynamic coefficients of the airfoils, which are lift, drag, and pressure coefficients, were calculated using the following equations, respectively.

$$C_L = \frac{F_L}{\frac{1}{2} \rho V_\infty^2 A} \quad (3)$$

$$C_D = \frac{F_D}{\frac{1}{2} \rho V_\infty^2 A} \quad (4)$$

$$C_P = \frac{P - P_\infty}{\frac{1}{2} \rho V_\infty^2} \quad (5)$$

In the equations, F_L is lift force, F_D is drag force, P is static pressure at the measured point on the wing, P_∞ is free flow static pressure, ρ is fluid density, V_∞ is free stream velocity, and A indicates wing reference area.

Leading edge deflection angle, its chordwise location and angle of attack were chosen as the main parameters. The schematic representation of the parameters is presented in the Figure 3.1. The deflection angle (θ) was defined downwards in all airfoils created. Kan et al.[60] found that the upward deflection angle decreases the maximum lift coefficient and the stall angle is negatively affected and the stall angle decreases due to the decrease in the negative pressure area and the flow separation at the trailing edge. Therefore, all modified airfoils are produced with a downward deflection.

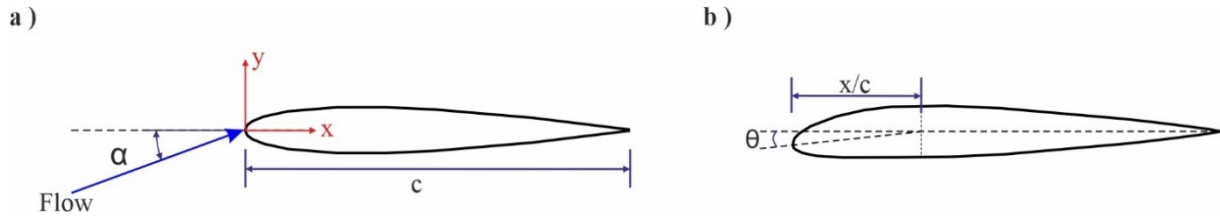


Figure 3. 1. Schematic representation of investigated parameters

Figure 3.2 shows the airfoil with 2° deflection and modifications up to 0.3 chord length imported into the ANSYS program. The airfoil with these criteria is named $\theta_D=2_x/c=0.3$. Figure 3.3 and Figure 3.4 show the airfoils with $\theta_D=4_x/c=0.3$ and $\theta_D=6_x/c=0.3$ respectively. The airfoils were analyzed under $Re=1 \times 10^5$ flow conditions after mesh structures were created in Fluent environment.



Figure 3. 2. Airfoil $\theta_D=2_x/c=0.3$



Figure 3. 3. Airfoil $\theta_D=4_x/c=0.3$



Figure 3. 4. Airfoil $\theta_D=6_x/c=0.3$

3.1. Generating Airfoil Geometry

The shape of deformation is parabolic as mentioned in the study of Kan et al.[60]. The trajectory equation of the mean camber line for the morphing wing is defined as follows;

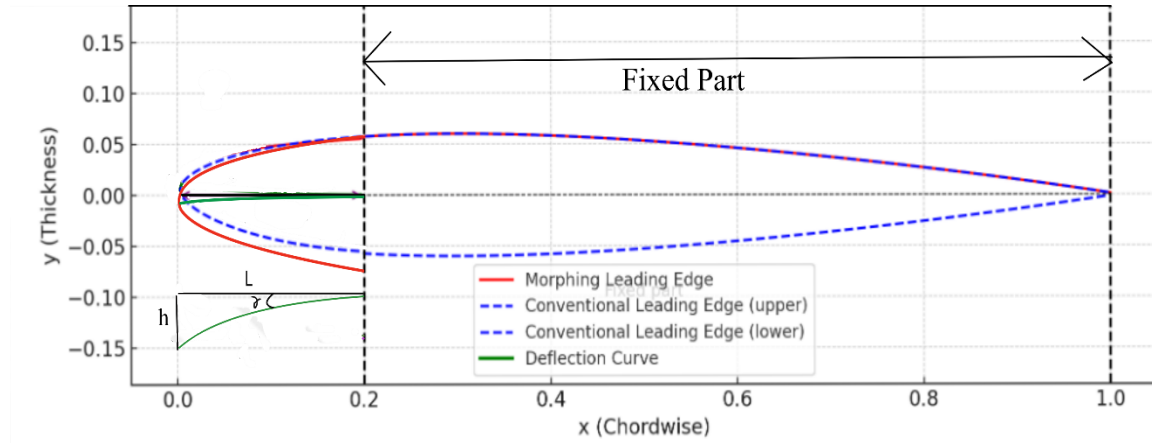


Figure 3. 5.Interior of deflection on leading edge of the airfoil

Since we are working on a symmetrical wing profile, the MCL (Mean Camber Line) trajectory equation can provide us with information about the deflection movement occurring along the leading edge. Typically, the MCL geometry of symmetrical wing profiles appears as a horizontal or nearly horizontal line however, for greater realism and ease of aerodynamic analysis, the MCL equation is represented as $y = a \cdot (x-0.2)^2$. Figure 3.5 illustrates the scenario where deflection begins at $x/c = 0.2$. In the versions where this situation starts from $0.1c$ and $0.3c$, 0.1 and 0.3 are subtracted from x in the expression inside the parentheses in the MCL equations, respectively. The starting point of the deflection is $(0.2, 0)$, and the tip coordinate of the leading edge of the deflected wing profile is $(0.2-dL, -dh)$. As a result of trigonometric equations, we can express $dL \approx \cos \gamma \times 0.2$. Therefore, the MCL equation appears as equation 8 below.

$$dL \approx \cos \gamma \times 0.2 \quad 6$$

$$\Delta h = \frac{-\tan \gamma}{0.2 \times \cos \gamma} (x-0.2) \quad 7$$

$$y = \frac{-\tan \gamma}{0.2 \times \cos \gamma} (x - 0.2)^2$$

8

The airfoil geometries are generated in the Matlab program using equations 6,7 and 8 with approximately 50 lines of code. The chord length of all created airfoil geometries is set as 0.12 m. Only the leading edge was modified for all test cases. The purpose here is to determine how a change in the leading edge changes the parameters such as lift, drag, turbulence and velocity distribution in the airfoil geometry. While creating the geometries, 2°, 4° and 6° were selected as deflection angles and for these three deflection angles, combinations were created at $x/c=0.1$, 0.2 and 0.3. For each case, analyzes were performed at 0°, 3°, 7°, 10°, 15° and 18° of attack angle.

Therefore, if the base case, that is, the case without deflection angle, is included, a total of 60 cases were created for the analysis environment. Figure 3.6 shows the geometries $\theta_D=4^\circ$, $x/c=0.1$, $\theta_D=4^\circ$, $x/c=0.2$ and $\theta_D=4^\circ$, $x/c=0.3$. Here, it can be observed how the change of x/c value at the same deflection angle (θ) affects the airfoil geometry. As can be seen, in the case of $x/c=0.3$, the pressure side of the leading edge is lower than the other cases. Figure 3.7 shows the geometries $\theta_D=2^\circ$, $x/c=0.3$, $\theta_D=4^\circ$, $x/c=0.3$ and $\theta_D=6^\circ$, $x/c=0.3$. Here, it can be observed how the airfoil geometries with different deflection angles will be formed for the same x/c value which is equal to 0.3. As can be noticed, the place where the deflection starts is the same in all three geometries, but in the geometry with more deflection (6°), the leading edge bends downwards more, while in the case where the deflection angle is the least (2°), the leading edge bends downwards minimally.

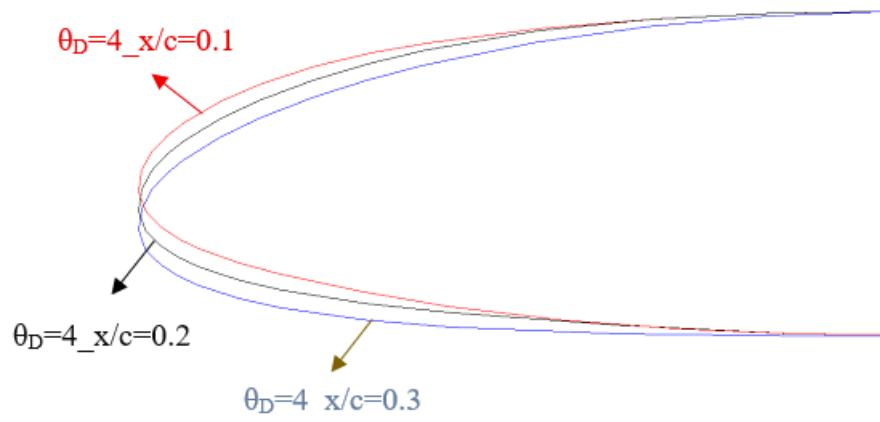


Figure 3. 6. Airfoils at 4° deflection

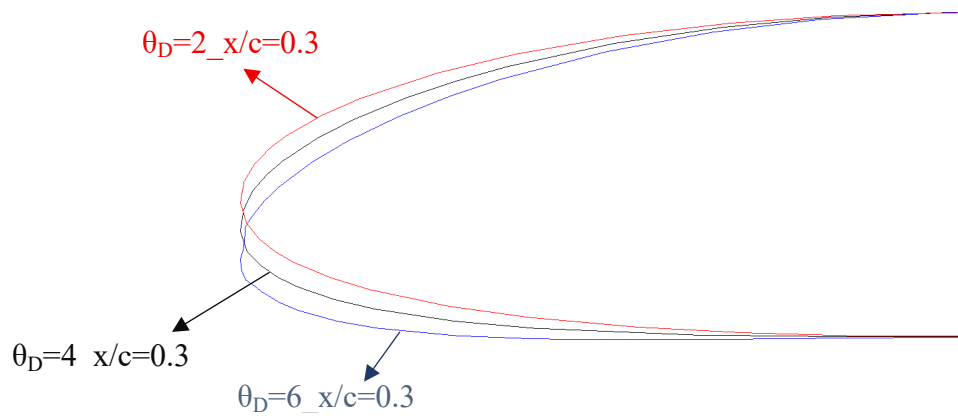


Figure 3. 7. A set of airfoils while x/c is 0.3

3.2. Computational Domain

For the NACA 0012 airfoil with a maximum thickness equal to 12% of the chord length in the base case, that is, without any deflection angle, a C type computational domain was created in the ANSYS Fluent package program. C type computational domain is one of the most frequently used geometries in airfoil studies. The reason for this geometry is that while in this geometry, a better mesh structure can be created from the regions close to the airfoil walls and more accurate sharp results can be obtained[86]. The computational domain is divided into two parts: airfoil and far field. The chord length of the airfoil is 120 mm, the rectangular sides of the solution domain are 3000 mm, and the radius of the circle is 1500 mm. These width and length values are sufficient to correctly solve the Navier Stokes (NS) equations loaded in the program's algorithm by taking advantage of the velocity fields in the flow. Excessive enlargement of the solution domain does not change the results of the parameters much, and since the number of mesh cells will increase with this area, the time required for the computer to reach the result will increase unnecessarily. While creating the solution domain, the domain is divided into certain regions by means of lines passing through certain points of the airfoil. These points are first the trailing edge point. The area to the right of this point has a completely rectangular geometry. The second point is the point corresponding to 30% of the chord length. This point was chosen because the thickness value generally reaches the maximum at this point. The third and last point is the leading edge point. While the solution domain is divided vertically at the first two points, it is divided horizontally at the third point.. Figure 3.8 shows the boundaries of the computational domain. The outermost part is far field and the geometry to be studied is assigned as airfoil.

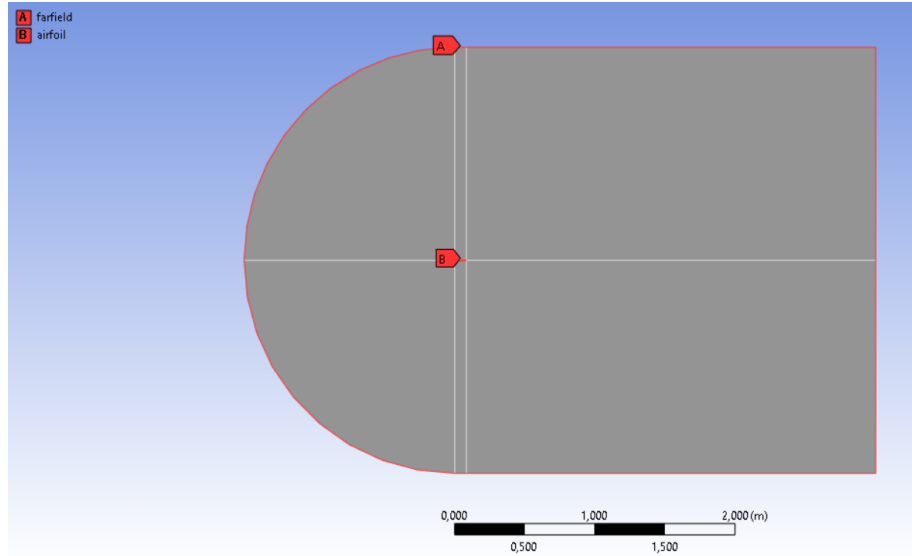


Figure 3. 8. Boundaries of the computational domain

3.3. Meshing

Mesh structures, which are much smaller in the solution domain, are one of the main components used in many engineering solutions, especially in aerodynamic problems. However, in order to achieve accurate and relatively satisfactory results, mesh structures must be created with appropriate techniques. In this study, as mentioned before, C mesh is used because in this structure, the mesh cells fit better with the airfoil wall and it is easier to create more refined mesh structures along the entire axis.[86] Apart from that, unstructured mesh structures can cause the lift coefficient to be higher than it should be, especially after 8° angle of attack, that is, at high angles of attack. Up to this angle of attack, structured and unstructured mesh structures give almost the same result.[87] For this reason, the solution area was separated from certain points to create structured mesh structures and lines separating these regions were assigned with the projection method. However, in cases where there are high lift devices such as flaps and slats, it is generally necessary to have a higher mesh count when calculating the drag coefficient in and around these regions of the wing due to the circulation and turbulence caused by the high lift force [86]. If there is less mesh density than necessary in these regions, the drag coefficient cannot be calculated correctly. In this study, since modifications are made on the leading edge, more mesh density is required in these regions. However, while applying

these mesh structures, it is necessary to pay attention to the y^+ value, which is a dimensionless parameter and helps to understand how good or bad the mesh is. Körpe et al.[88] examined the flow performance of a non-symmetrical airfoil with values starting from 1 and going up to 105 for y^+ . They noticed that as this value decreases, the separation region diagnosed on the airfoil approaches the trailing edge. This was visualized with shear stress pathlines graphs. It was observed that the drag coefficient, especially at high angles of attack, does not converge and reaches very high values if the y^+ value is high. For this reason, the y^+ value was tried to be kept between 1 and 2 with the mesh refinement processes. Figure 3.9 shows the y^+ plot of the NACA 0012 airfoil for the $\theta_D=6^\circ$, $x/c=0.3$ model after the meshing process is finished. The total number of mesh elements in the solution domain was found to be 124800 and the number of nodes was found to be 125760. Face meshing was applied to achieve a smoother mesh structure along the entire surface. There is also a table with mesh quality values in Table 3.1. According to the recommendations of ANSYS, the maximum Skewness Quality value between 0.5-0.8 indicates a good mesh. Likewise, according to the minimum Orthogonal Quality value, we understand that there is a good mesh structure. Figure 3.10 shows the mesh structure inside the computational domain after meshing.

Table 3. 1. Mesh quality values

	Minimum	Maximum	Average	Standard Deviation
Element Quality	$6,98 \times 10^3$	0,99947	0,42257	0,30997
Skewness Quality	$1,31 \times 10^{-10}$	0,59844	$5,18 \times 10^{-2}$	$7,94 \times 10^{-2}$
Orthogonal Quality	0,57349	1	0,98904	$3,90 \times 10^{-2}$

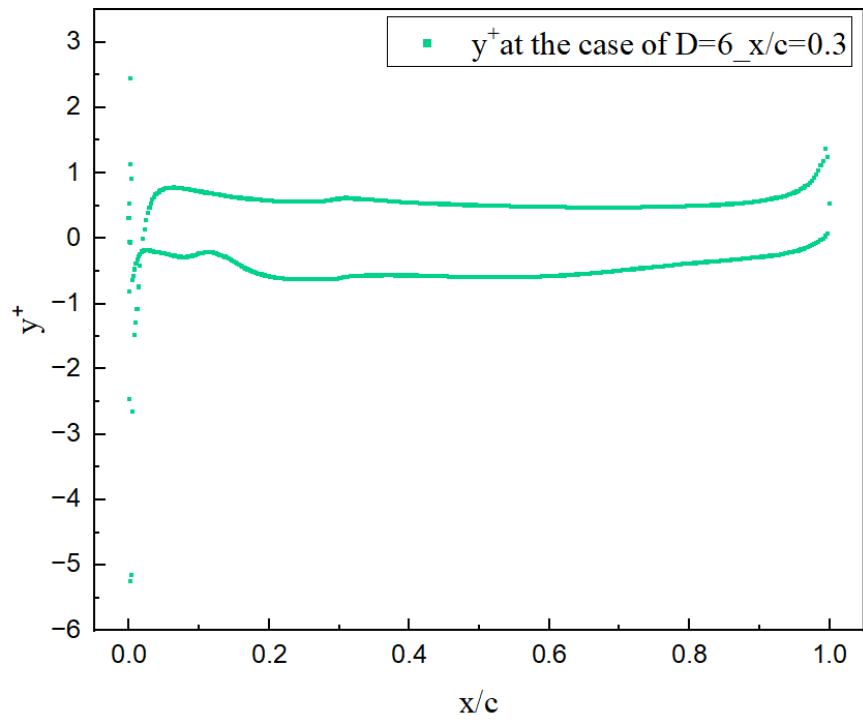


Figure 3. 9. Y^+ graph of the airfoil for one solution case

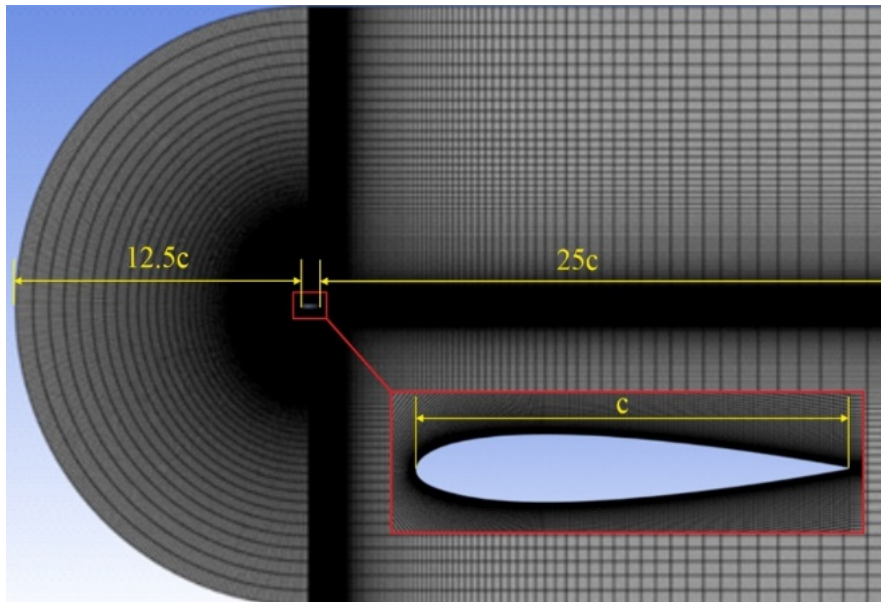


Figure 3. 10. Mesh all around the computational domain

3.4. Solution Setup

After the mesh structure of each airfoil was created, density based, and steady flow was selected as the solver type in Fluent program. Transition k-kl-omega(3eqn) was selected as the turbulence model. This model was preferred because it makes turbulence modeling from the inlet more clearly and is frequently used in such problems. This model uses basically 3 equations to model turbulence including the Reynolds Averaged Navier Stokes (RANS) equations as given below. Equation 1 is the continuity equation. If the flow is incompressible, this equation must be equal to 0. Equation 2 is the momentum equation, where $\vec{\tau}$ is the stress tensor and $\rho\vec{g}$ is the gravitational force. The equation 3 is the turbulence kinetic energy (k) equation, where the P_k term is the generative turbulence kinetic energy, the μ_t term is the turbulence viscosity and the β^* and σ_k are the model constants.

$$\frac{\partial \rho}{\partial t} + \frac{\partial(\rho u_j)}{\partial x_j} = 0 \quad (1)$$

$$\frac{\partial}{\partial t}(\rho \vec{v}) + \nabla \cdot (\rho \vec{v} \vec{v}) = -\nabla p + \nabla \cdot (\vec{\tau}) + \rho \vec{g} + \vec{F} \quad (2)$$

$$\frac{\partial(\rho k)}{\partial t} + \frac{\partial(\rho k u_j)}{\partial x_j} = P_k - \beta^* \rho k \omega + \frac{\partial}{\partial x_j} \left[(\mu + \sigma_k \mu_t) \frac{\partial k}{\partial x_j} \right] \quad (3)$$

Air is used as the flow material and is entered into the system as an ideal gas. Gauge pressure is used as 101325 Pa. Since the Reynolds number is used as 1×10^5 , the velocity value at this number is 12.67266 m/s. All analyses were carried out with steady state analysis. In this stage, coupled- implicit scheme and density based solver were used. The least square cell-based method was used for the gradient and the second order discretization method was applied for all other parameters in the spatial discretization. 10^{-6} is chosen for the residual convergence monitor. In this way, the residuals get close enough to each other and the results are obtained after the solution becomes relatively stable. Normally, the flow analysis of each modified airfoil was performed in Fluent program at $0^\circ, 3^\circ, 7^\circ, 10^\circ, 15^\circ$ and 18° angle of attack, but since it would

be both very laborious and very time consuming to rotate each airfoil at the appropriate angle of attack and produce mesh structures, instead of this, the mesh structure of the airfoil at 0° angle of attack was produced and the component effects of the incoming flow in the x and y directions were entered into the system in the boundary conditions. After the velocity values are entered, the computational domain is relatively large and the solution time is a bit high due to the number of mesh cells. Each solution takes almost 5-6 hours and in total 60 solutions were made, including the base case.

3.5. Mesh Independence

During the main work to be done after performing mesh processing, a mesh independence study was performed to determine the optimal number of mesh cells. This study is usually done by starting with a small number of meshes and gradually increasing this number and obtaining the results and comparing them with each other. Since the distances between the cells will be shorter at higher mesh numbers, the Navier Stokes equations solved by the program are more precise and accurate. Of course, the disadvantage of this is that it is reflected as a serious loss of time. Thanks to the mesh independence study, the number of cells where the difference between the performance parameters does not change much is selected and both relatively accurate results are obtained and a lot of loss of time is prevented. In this sense, this study can be defined as an optimization process. For this purpose, the lift coefficient was first obtained using a mesh cell number of 59400. Then the same process was performed for mesh numbers 124800 and 238000. Finally, the change in the C_L for these three cases was observed. Figure 3.11 shows the values of the lift coefficient for 3° , 7° , 10° and 15° angle of attack in these three cases. It is noted that the change in the C_L decreases significantly after the mesh cell number 124800. Therefore, the mesh number to be used in the analysis was selected as 124800.

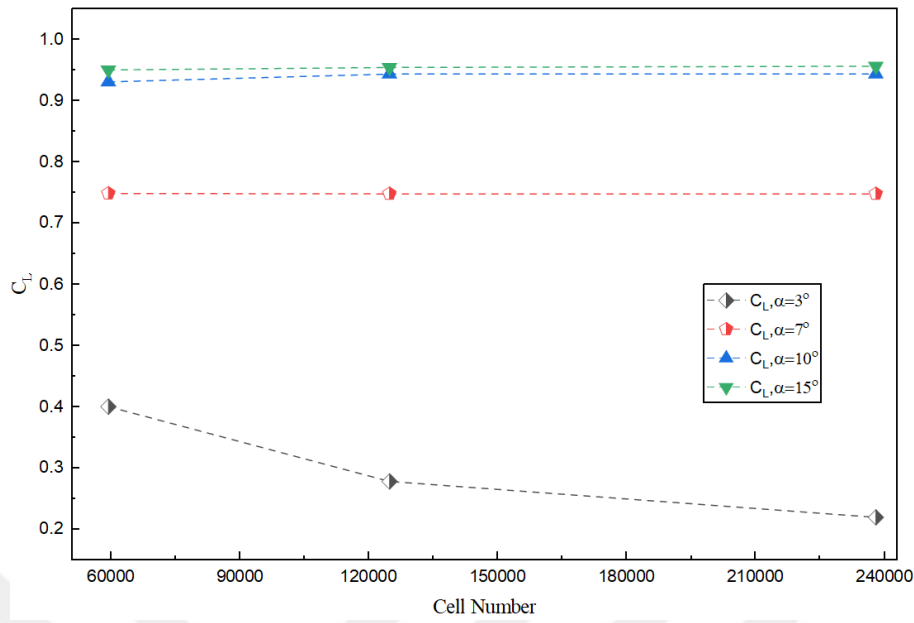


Figure 3. 11. Mesh Independence study at four different angles of attack

3.6. Validation

Before starting the individual study, validation was carried out with the help of sources that have been previously studied close to this field. The aim here is to obtain computational performance data under the same conditions, provided that the values do not deviate too much from the tolerance values. For this purpose, Zakaria et al.[89] studied the vortex formation of the NACA 0012 airfoil in general at high angles of attack and performance data with various solutions. They compared the Spalart Allmaras (SA), Laminar, Reynolds Stress Model (RSM) at $Re=79900$ with previous experimental results. In the individual study, in order not to increase the intensity of the future analysis too much, the data from the angle of attack from 0° to 40° were added to the graph. Here, the C_L and C_D graphs obtained in the SA solver were compared with the graphs obtained individually. The resulting graph, which contains the comparison of all the data, is given in Figure 3.12. As can be clearly seen in the figure, the data in the individual study generally falls between the simulation and experimental data in the previous study and does not go beyond the 10% tolerance values. Therefore, the validation process has been successfully completed.

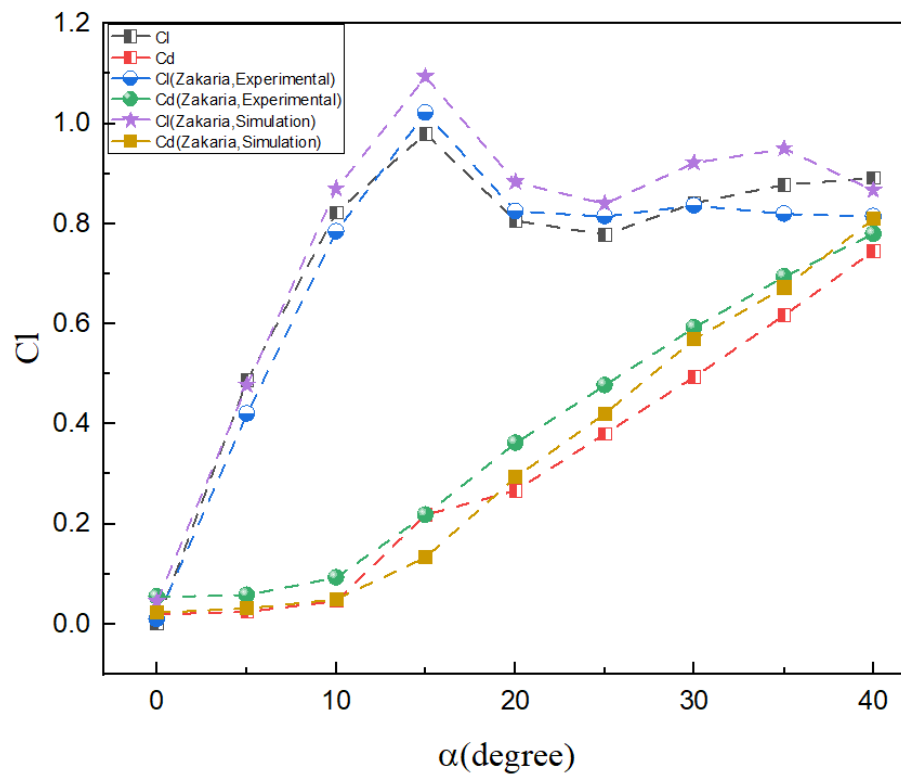


Figure 3. 12. Validation Results

4. RESULTS AND DISCUSSION

All modified NACA 0012 airfoils, by defining deflection angles ($\theta = 2^\circ, 4^\circ$, and 6°) with initiated at $0.1c$, $0.2c$, and $0.3c$, were compared with the base case. Therefore, the lift coefficient (C_L), drag coefficient (C_D), and C_L / C_D (aerodynamic efficiency) results are first presented to reveal the effect of deflection angles as well as its starting points. To define the optimum airfoil for each angle of attack, the percentage differences of C_L and C_D the is of great importance from flow control perspective. In this sense, pressure coefficients (C_p) were highlighted for different angles of attack for the comparison of its change with respect to streamwise direction. Hence, it can be observed how increasing angles of attack affect the aerodynamics performance of the airfoils. Also, streamwise velocity contours and turbulent kinetic energy (TKE) contours for all deflected angles were presented to understand how the flow is manipulated through downstream of the airfoils with the help of the deflection at various locations of the base airfoil. Lastly, the average streamlines are demonstrated to underline the flow topology specifically for the flow control situations.

Therefore, due to the longer flow path over the upper surface of the wing, the flow velocity is higher and the pressure is lower, in accordance with Bernoulli's principle. The low-pressure regions observed in the pressure contour plots are typically associated with flow acceleration. However, if the flow separates from the airfoil surface or becomes unstable, pressure on the surface may further decrease due to the presence of recirculating or unsteady flow structures. This pressure reduction can be either beneficial or detrimental, depending on the location. If it occurs on the pressure side (lower surface), the pressure difference between the upper and lower surfaces decreases, resulting in a lower lift coefficient. Conversely, if the pressure drop occurs on the suction side (upper surface), the lift may increase due to a greater pressure differential. However, if this effect becomes excessive and leads to significant flow separation, the airfoil may enter a stall condition, causing a sudden and substantial drop in lift.

Figure 4.1 presents the C_L and C_D values for various angles of attack for base and 2° deflected situations. From 0° to 18° angles of attack ($\Rightarrow$), the C_L values of the airfoils are shown on the left side, while the C_D values are shown on the right side of this figure. At $\Rightarrow=0^\circ$, the airfoil of $\theta_D=2^\circ$ $x/c=0.1$ can generate a limited lift compared to base case, but the other modified airfoils produce negative lift. The reason for this can be explained by the fact that the flow separates

from the surface and the lift decreases due to the increase in curvature in the suction part of the leading edge, because the momentum of the flow is low due to the low attack angle. In the following sections, dimensionless velocity and turbulence kinetic energy (TKE) contour graphs of all airfoils will be presented. Especially in the velocity contour graphs, if the pressure is high in the regions with high camber, it should be understood that the velocity of the flow decreases and stagnates in these regions and exerts a downward force on the airfoil, on the contrary, if the flow accelerates in these regions and the pressure decreases, it should be understood that the flow is attached to the surface in these regions. In other words, we can talk about a laminar formation in the boundary layer. At $\alpha=3^\circ$, almost all airfoils produce the same lift force except for $\theta_D=2_x/c=0.3$. Similar magnitudes of C_L can be seen at $\alpha=7^\circ$. With the increasing angles of attack, the airfoil of $\theta_D=2_x/c=0.3$ produces 4.88% higher lift than the base case at $\alpha=10^\circ$. The same airfoil profile produced 7.86% less drag than the base model at that angle of attack. Although the base condition showed a certain amount of lift increase after a 10° angle of attack, models other than the $\theta_D=2_x/c=0.1$ model slowly transitioned to a stall condition after this angle. At $\alpha=18^\circ$, the base airfoil produces 1.1% more lift than the $\theta_D=2_x/c=0.3$ airfoil because this model transitioned to stall at 10° angle of attack. In the $\theta_D=2_x/c=0.1$ model, a slight increase in lift was observed after a 10° angle of attack, similar to the base condition. When C_D variation is analyzed, the difference between the base airfoil and the modified airfoils widens at $\alpha \geq 10^\circ$. C_D of modified airfoils slightly lower than base case, indicating higher aerodynamic performance for the modified airfoils. Especially, at $\alpha=10^\circ$, $\theta_D=2_x/c=0.3$ airfoil produces 30.19% less drag than the base airfoil.

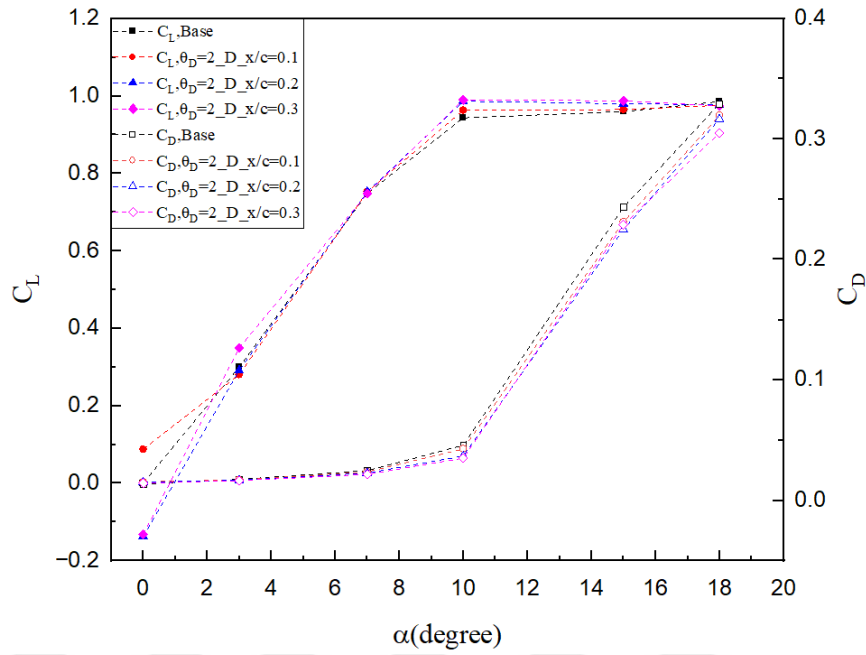


Figure 4. 1. C_L and C_D of Base and 2° deflected modified airfoils

Figure 4.2 shows the C_L and C_D values of the base case and the modified airfoils with the deflection of $\nabla=4^\circ$. The biggest difference in lift coefficient with the base case occurred at 10° angle of attack. All modified airfoils of $\nabla=4^\circ$ reveal more lift coefficient than the base case at this angle of attack. Also, $\theta_D=4_x/c=0.3$ airfoil produced 8% higher lift relative to the base case. For drag coefficient, all airfoils produce similar drag up to $\Rightarrow=10^\circ$ and at this angle of attack, the airfoil model $\theta_D=4_x/c=0.3$ produced 37.75% less drag, additionally it has 12.6% less drag at $\Rightarrow=18^\circ$ compared to base airfoil. The $\theta_D=4_x/c=0.1$ model continued to increase lift up to an angle of attack of 18° , similar to the base model. The $\theta_D=4_x/c=0.2$ model stalled after 10° angle of attack. The $\theta_D=4_x/c=0.3$ model stalled after 15° of angle of attack. Therefore, models with this deflection angle exhibited lower stall performance compared to the base model. This is because the base model continues to increase lift up to an angle of attack of 18° . However, despite this, $\theta_D=4_x/c=0.2$ produced 3.9% more lift than the base model in 15° angle of attack or in the post-stall region.

Figure 4.3 demonstrates the C_L and C_D values of the base case and the modified airfoils with $\nabla=6^\circ$ deflections, which is the largest deflection angle in this study. The $\theta_D=6_x/c=0.2$ airfoil has the biggest difference in lift compared to the base case. It produced a 5.4% higher lift

compared to the base case at 10° angle of attack. The $\theta_D=6_x/c=0.1$ and $\theta_D=6_x/c=0.2$ models stall at an angle of attack of 15° , while the $\theta_D=6_x/c=0.3$ model continues to increase lift up to an angle of attack of 18° , similar to the base condition, and this model produces 7.27% more lift at an angle of attack of 3° compared to the base condition. The lowest drag coefficient compared to the base condition was observed at a 10° angle of attack and in the $\theta_D=6_x/c=0.3$ model. This model produced 32.87% less drag compared to the base condition. Based on this informations, only the $\theta_D=2_x/c=0.1$ and $\theta_D=6_x/c=0.3$ models showed lift increase up to an attack angle of 18° , similar to the base condition. At low angles of attack such as 3° and intermediate angles such as 7° to 10° , the $\theta_D=4$ models produced lower drag, while at high angles such as 15° to 18° , the $\theta_D=6$ models produced lower drag. In terms of lift coefficient, at angles of 3° to 7° , the $\theta_D=2_x/c=0.3$ and $\theta_D=2_x/c=0.2$ models produced the highest lift compared to the base condition respectively. Thus, at small attack angles, the $\theta_D=2$ models performed better. At attack angles of 10° , 15° , and 18° , the models that produced the highest lift compared to the base condition were $\theta_D=4_x/c=0.3$, $\theta_D=6_x/c=0.2$, and $\theta_D=6_x/c=0.3$, respectively. It is also noteworthy that at a high angle of attack of 18° , the model $\theta_D=6_x/c=0.3$ produces 18.37% less drag than the base model.

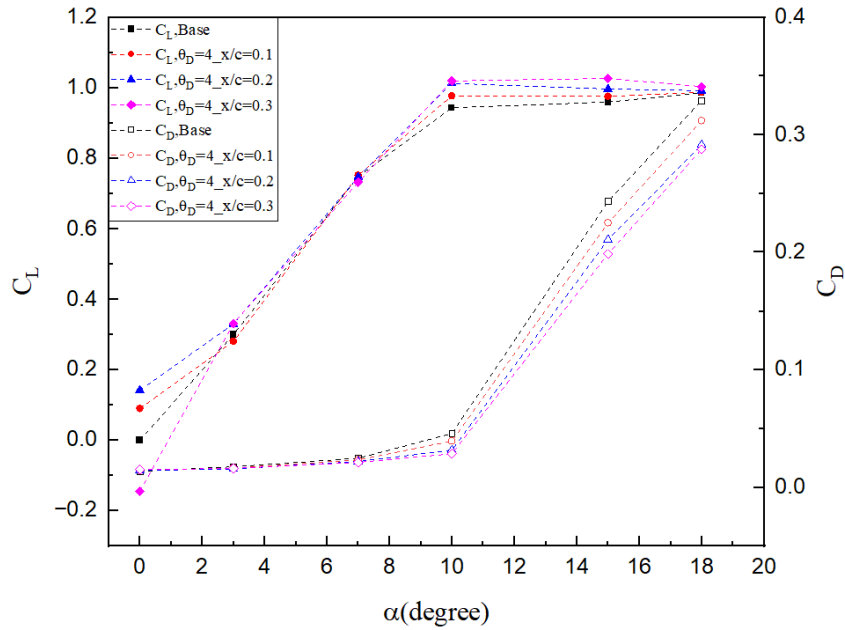


Figure 4. 2. C_L and C_D of Base and 4° deflected modified airfoils

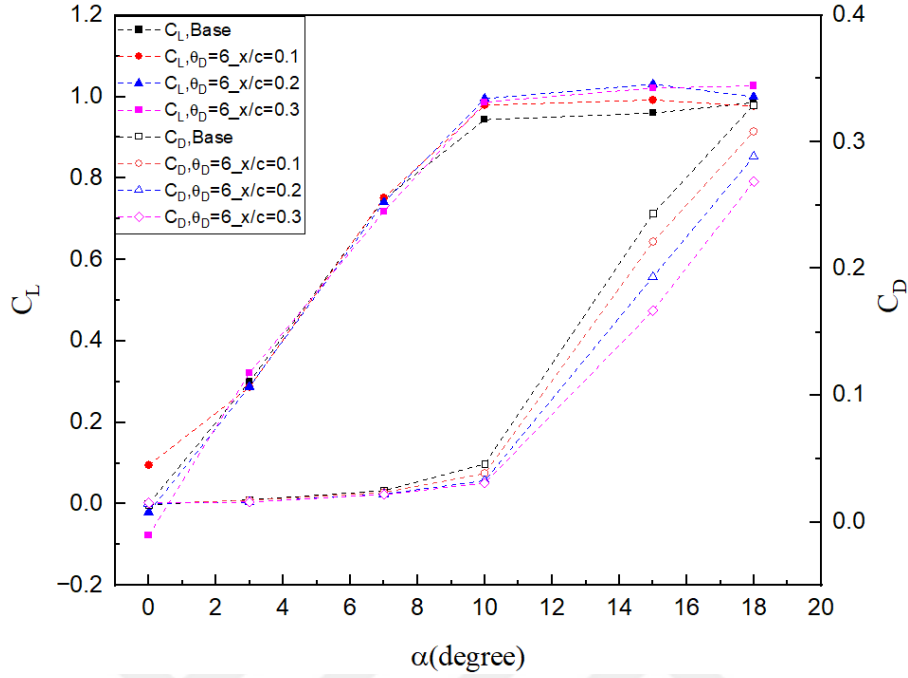


Figure 4. 3. C_L and C_D of Base and 6° deflected modified airfoils

However, the morphing models proved to be a very effective method of reducing the drag coefficient.

The aerodynamic efficiency (C_L/C_D) is a ratio that gives the relationship between the lift and drag coefficients of an aerodynamic aerial profile and is frequently used in performance analysis. Figure 4.4 presents the C_L/C_D graph of the base case and the modified airfoils with 2° deflection angles at angles of attack from 0° to 18°. For this deflection situation, the lift to drag ratio of all airfoils is almost at the same magnitudes for 15° and 18° angles of attack. Since $\theta_D=2_x/c=0.2$ and $\theta_D=2_x/c=0.3$ airfoils produced negative lift at $\alpha=0^\circ$, the lift to drag ratio was negative. The biggest difference to the base case occurred in stall angle which is at $\alpha=10^\circ$ for $\theta_D=2_x/c=0.3$. Since this airfoil profile stalls at 10° , the increase in aerodynamic efficiency in the post-stall region compared to the base condition can be considered very good. At this angle, the $\theta_D=2_x/c=0.3$ airfoil had a 36.55% higher C_L/C_D ratio than the base airfoil. This higher ratio was due to a considerable decrease in drag compared to the base case. At $\alpha=18^\circ$, the same airfoil has a 6.74% higher C_L/C_D ratio compared to the base airfoil. Specifically, all

modified airfoils display higher aerodynamic performance relative to base case at $\alpha=7^\circ$ and 10° .

In the pre-stall region, such as 3° , the $\theta_D=2_x/c=0.3$ model again had the best aerodynamic efficiency value. It had a 24.5% higher efficiency value compared to the base condition. At this angle, all models except the $\theta_D=2_x/c=0.1$ model exhibit higher efficiency than the base model. Only the $\theta_D=2_x/c=0.1$ model has a 4.19% lower efficiency compared to the base condition. At high angles of attack, all models have higher efficiency than the base condition.

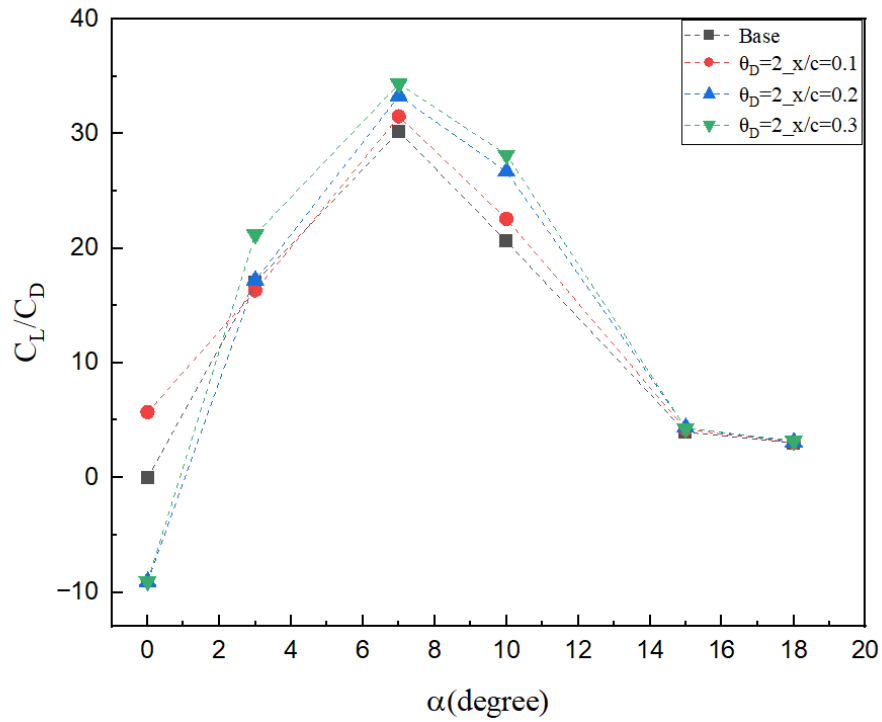


Figure 4. 4. C_L/C_D Graph of Base and 2° deflected modified airfoils

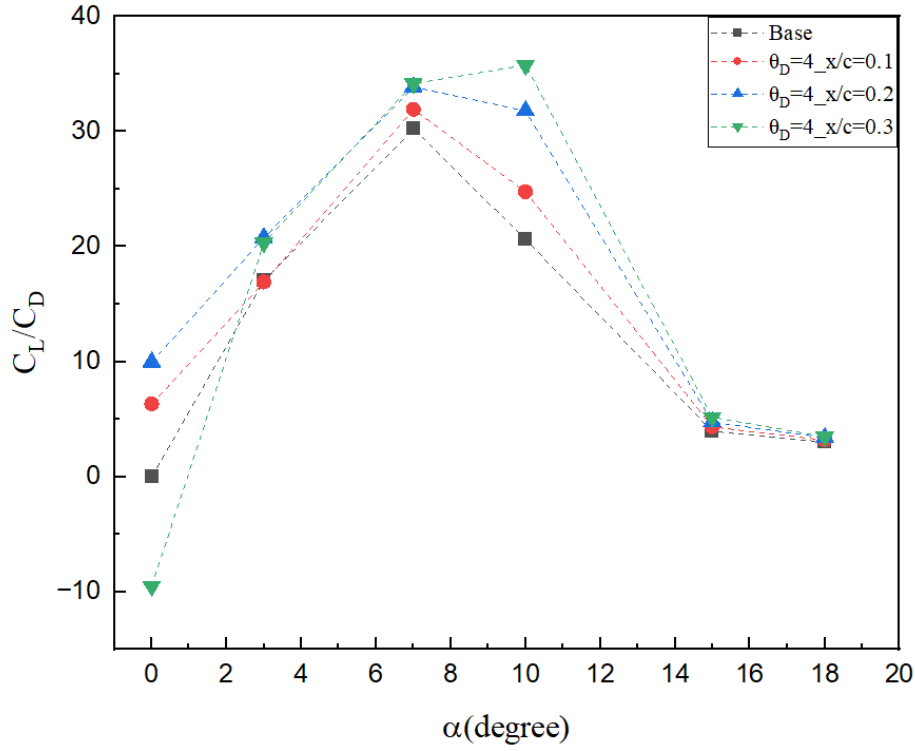


Figure 4. 5. C_L/C_D Graph of Base and 4° deflected modified airfoils

Figure 4.5 illustrates C_L/C_D ratios for the base case and the airfoils with modification of 4° deflection. Unlike the 2° deflected model, $\theta_D=4_x/c=0.2$ airfoil also produces positive lift at $\Rightarrow=0^\circ$, therefore, the aerodynamic efficiency value is positive. Up to $\Rightarrow=7^\circ$, the C_L/C_D ratio of the $\theta_D=4_x/c=0.2$ airfoil produced the highest difference compared to the base airfoil, but after this angle of attack, the $\theta_D=4_x/c=0.3$ airfoil showed the biggest difference compared to the base case. Since this model stalls at 15° , it achieves 16.4% higher efficiency at a post-stall angle of 18° compared to the base condition. Considering the 19.3% increase in efficiency at a pre-stall angle of 3° , the highest efficiency is achieved at medium angles of around 10° . $\theta_D=4_x/c=0.3$ airfoil has a 73.61% higher C_L/C_D ratio at $\Rightarrow=10^\circ$ compared to the base case. At $\Rightarrow=15^\circ$, the airfoil $\theta_D=4_x/c=0.3$ with excessive increasing drag produced a 31.1% higher C_L/C_D ratio compared to the base case. As can be seen, with an increasing deflection angle, higher aerodynamic efficiency can be achieved at high angles of attack like 15° or 18° compared to the base condition.

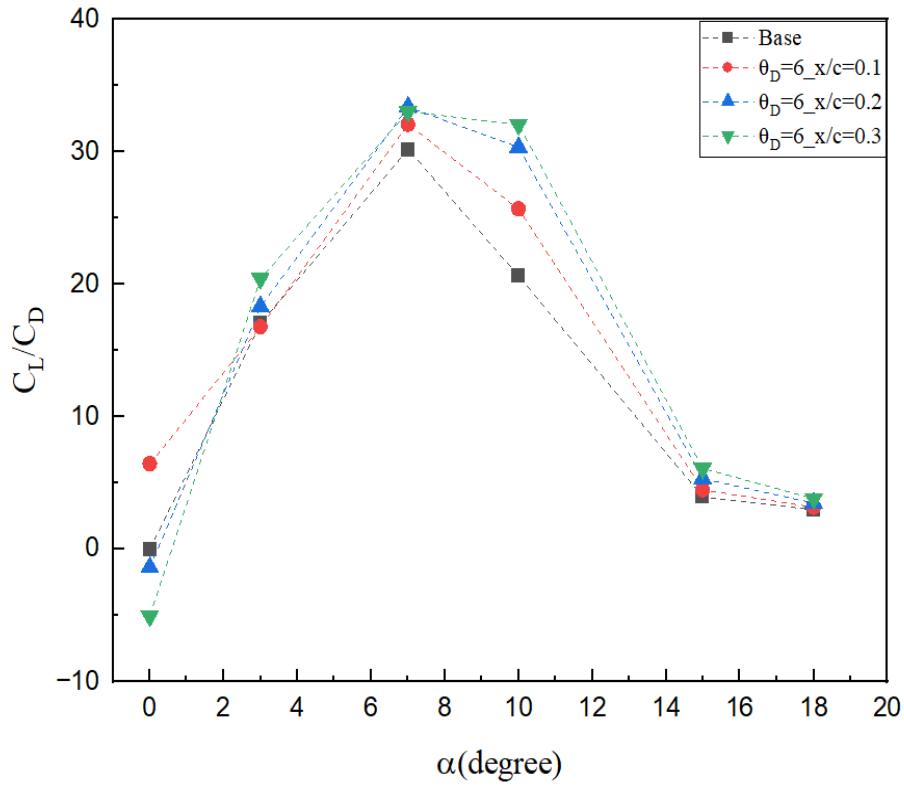


Figure 4. 6. C_L/C_D Graph of Base and 6° deflected modified airfoils

Figure 4.6 shows the plot of the C_L/C_D for airfoils with modifications at 6° deflection angle. In this case, the airfoil with modification of $\theta_D=6_x/c=0.3$ has the highest aerodynamic efficiency ratio compared to the base case at $\alpha \geq 10^\circ$. At 10° angle of attack, the airfoil $\theta_D=6_x/c=0.3$ has 55.6% higher aerodynamic efficiency compared to the base airfoil. At $\alpha=18^\circ$, the same airfoil had 27.6% higher aerodynamic efficiency compared to the base case. In both deflection cases, the curve of C_L/C_D becomes almost stable at $\alpha \geq 15^\circ$. This implies that the flow goes into a stall state yielding lift is in unchanged situation with the increase in drag. Also, considering all other cases with modifications, the airfoils with the highest aerodynamic efficiency compared to the base case in the post-stall region such as $\alpha=18^\circ$ were the airfoils with $\theta=6^\circ$ deflection. However, at transition angles of attack such as $\alpha=10^\circ$, the highest aerodynamic efficiency compared to the base case was observed in the models with $\theta=4^\circ$ deflections.

By plotting the pressure coefficient (C_p) distributions of the airfoils along the chord, it can be understood how the flow affects the airfoils for each of them. For six different angles of attack, $-C_p$ plots were created, and base and modified airfoils were compared with respect to changing

chord lengths, x/D . Figure 4.7 shows the $-C_p$ distributions of base and modified airfoils with all deflection angles for $\alpha=0^\circ$. In these figures, the area between the lower curve and the upper curve represents the lift produced and the upper curve shows the suction part of the airfoils, and the lower curve shows the pressure part. The first noticeable point is the amount of lift produced around the leading edge when the deflection angle increases from 2° to 6° . Since there will be more curvature on the leading edge at 6° deflection, the dynamic pressure of the flow increases and the static pressure decreases. Therefore, in this region, the negative pressure value in the suction region of the airfoil decreases even more, but as it moves towards the trailing edge, the upper and lower surfaces get closer to each other. The reason for this is most likely due to the high momentum of the accelerated flow, which cannot hold on to the surface due to its high momentum. In other words, the modified airfoils with lower deflection produce more lift towards the trailing edge. Also, similar for all airfoils, oscillations occurred in the regions close

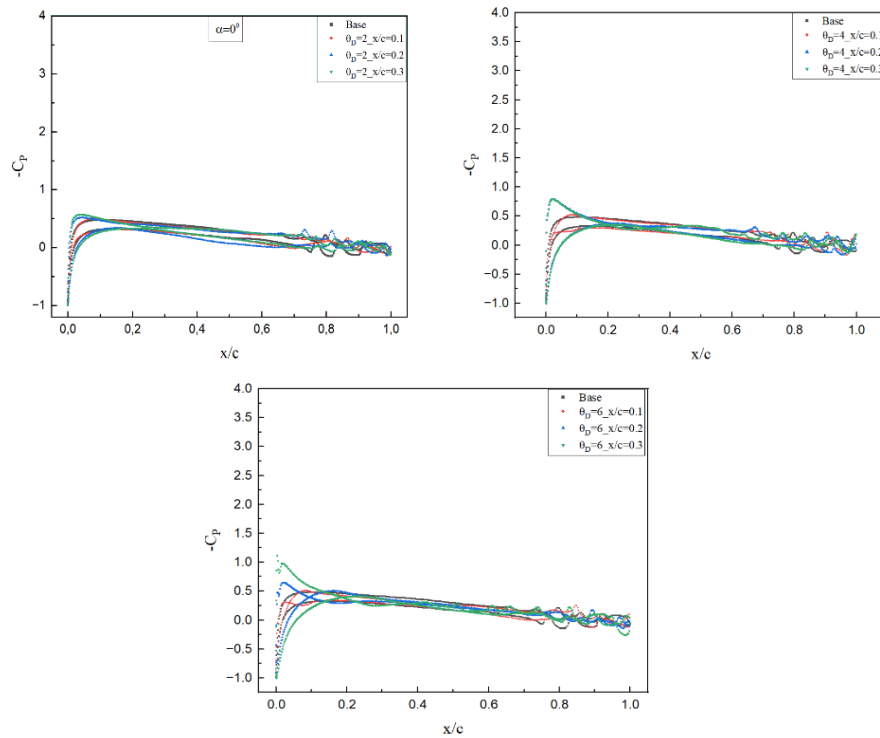


Figure 4.7. $-C_p$ Graphs of Base and all modified airfoils at $\alpha=0^\circ$

to the trailing edge due to the separation at this location. Figure 4.8 shows the $-C_p$ plots of all airfoils at $\alpha=3^\circ$. It is seen that the negative pressure on the leading edge increases as the

deflection angle increases because of the leading edge separation. The base airfoil showed the best performance in this field. Like $\alpha=0^\circ$, oscillations were observed close to the trailing edge, but their intensity decreased to a certain extent. Again, due to the low angle of attack and the lack of sufficient energy of the flow, the values between the two curves were close to each other.

Figure 4.9 shows the $-C_p$ graphs of all airfoils at $\alpha=7^\circ$. As can be noticed, while the peak point of $-C_p$ was around 1 in the previous situations, it increased to around 2.5 at this angle of attack, as a sign of increased lift. Laminar Separation Bubble (LSB) starting position can be observed from this figure. The concave part in the upper curves appears due to the presence of LSB. Additionally, the point where the bubble starts is called the separation point. The parts where the bubble expands are called the transition part. The point where it connects to the upper surface is called the reattachment point. The separation point of the base airfoil is approximately $0.08c$, whereas it is $0.28c$ for the $\theta_D=4^\circ$ $x/c=0.3$ airfoil, which is a positive effect from an aerodynamic perspective. Furthermore, unlike the cases at $\alpha=3^\circ$, trailing edge separations are almost eliminated at this angle of attack. Figure 4.10 shows the $-C_p$ plots of the base and all the modified airfoils for $\alpha=10^\circ$. With increasing the angle of attack, the peak point of $-C_p$ magnitudes reached to around 3.5. As in the previous angle of attack, the LSB is located at certain positions of the airfoils, but their intensity has changed. Also, in the modified airfoils, the LSB is located closer to the trailing edge compared to the base model. At this angle of attack, the separation and reattachment points of the airfoils with $\theta=2^\circ$ deflection angles and the base airfoil are almost the same. Therefore, it can be stated that the lift generated should be close to each other. In the airfoils with $\theta=4^\circ$ deflection, the regions where LSB occurs are quite different from each other. The first separation point of the LSB occurred at the base airfoil, while the last separation point occurred at $\theta_D=4^\circ$ $x/c=0.3$ airfoil. The first reattachment point occurred at the base airfoil and the last reattachment point occurred at $\theta_D=4^\circ$ $x/c=0.3$ airfoil. However, the transition distance in terms of chord length increases from the base airfoil to the modified airfoils. At $\theta=6^\circ$ deflection angle, the flow transitions more smoothly than in the other cases.

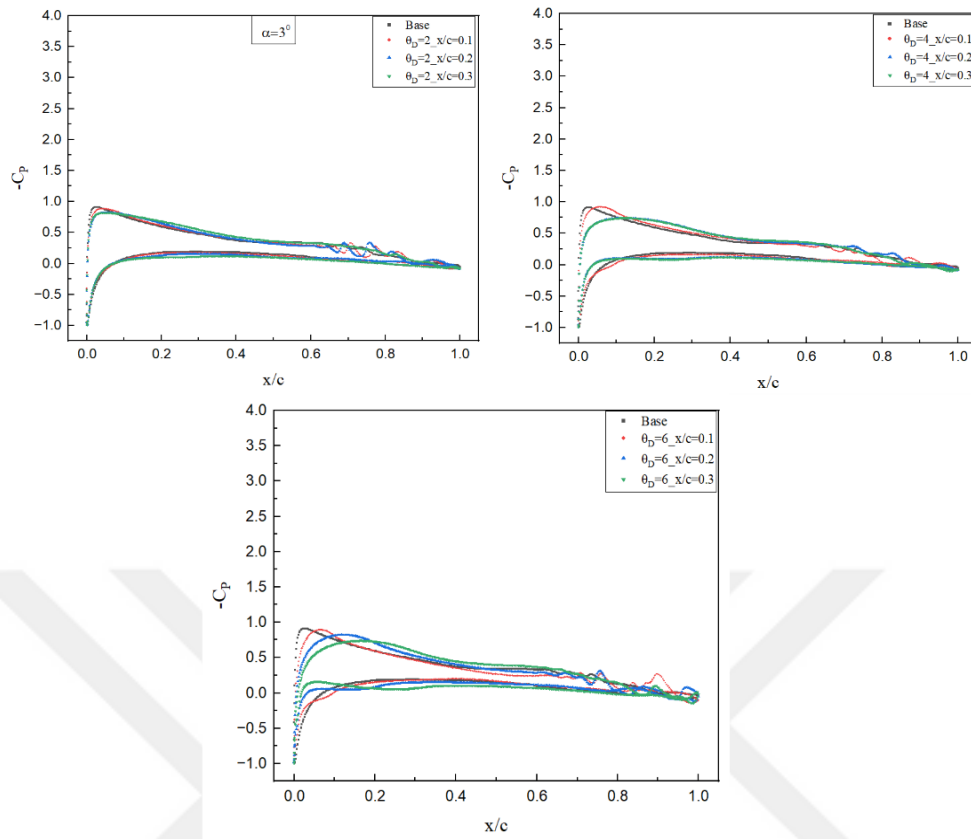


Figure 4. 8. $-C_p$ Graphs of Base and all modified airfoils at $\alpha=3^\circ$

Figure 4.11 shows the $-C_p$ plots of the base and all modified airfoils for $\alpha=15^\circ$. Compared to previous angle of attack, the peak value of $-C_p$ decreases from 3.6 to 2. This indicates that the airfoils enter stall condition. Unlike other angles of attack, the pressure side part of the airfoils with a deflection angle of $\theta=6^\circ$ is positively increased compared to the base model. In models with low deflection angles such as $\theta=2^\circ$, the pressure side curves are almost the same as the base model. Because of higher angle of attack, flow separations in the pressure side reduce the pressure and narrow the area of the C_p distribution. Modified airfoils may have increased the pressure by reattaching flow to the surface in these parts. Therefore, the lift is positively affected, and aerodynamic performance is improved. With the result obtained here, it can be interpreted that the modified airfoils with high deflection at higher angles of attack increase the pressure and the aerodynamic performance.

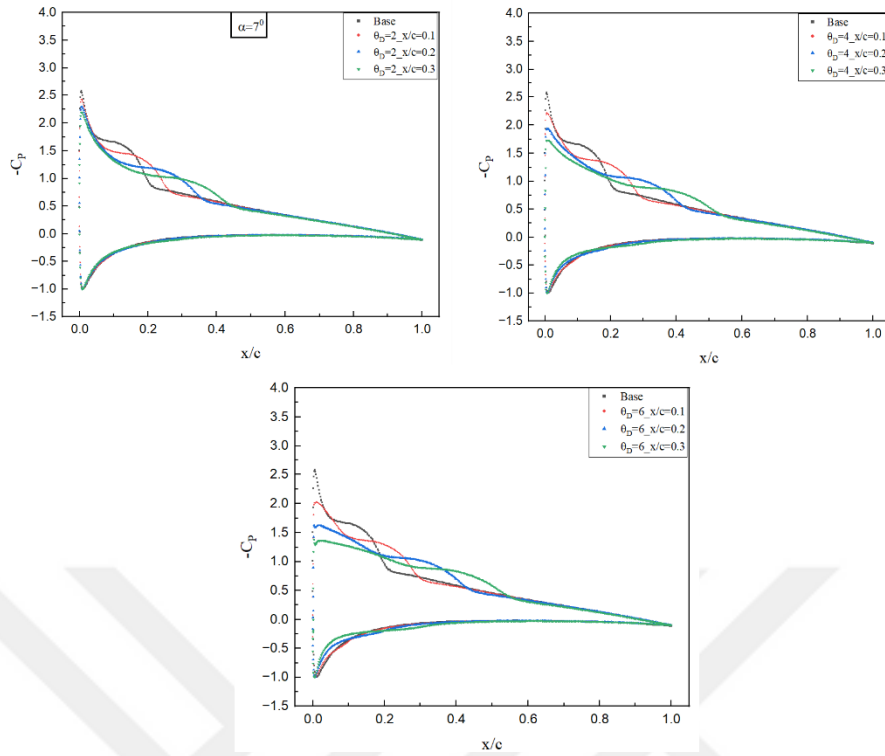


Figure 4. 9. $-C_p$ Graphs of Base and all modified airfoils at $\alpha=7^\circ$

Figure 4.12 shows the $-C_p$ plots of base and all modified airfoils for $\alpha=18^\circ$. At this angle of attack, all airfoils have now in the post stall state, which can be seen from the decrease in the magnitudes of the $-C_p$. The peak pressure coefficient, which was around 2 in the previous case, has decreased to 1.5. In these flow conditions, all the modified airfoils are quite different from the base case both on the pressure side and suction side of the airfoils. All modified airfoils showed better aerodynamic performance than the base model at this angle of attack. In the light of all these results, at $\alpha=0^\circ$, the deflection angle decreases the pressure on the upper surface of the wing and the distribution trend moves upward. On the contrary, at $\alpha=3^\circ$, the pressure distribution remained higher in the base model due to the leading edge flow separation.

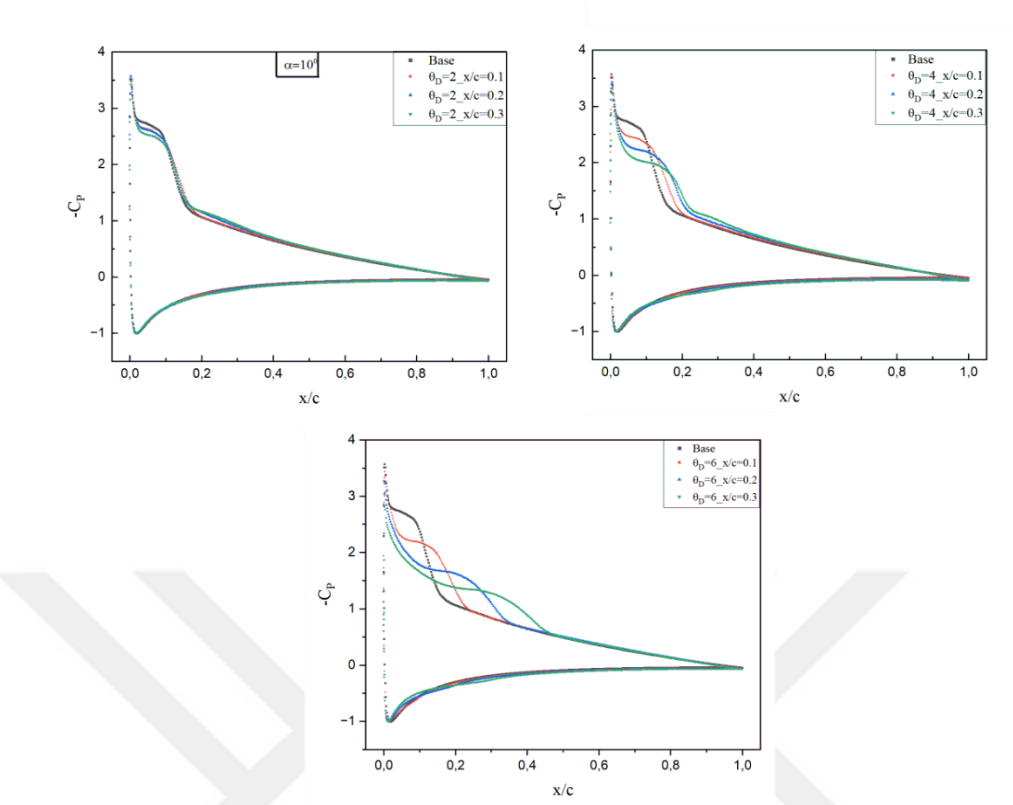


Figure 4. 10. $-C_p$ Graphs of Base and all modified airfoils at $\alpha=10^\circ$

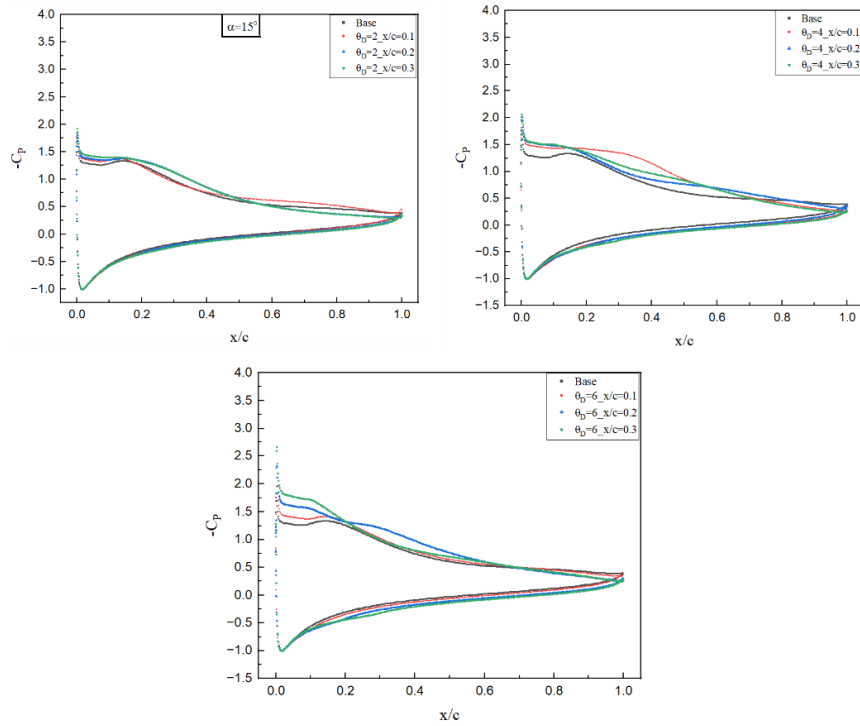


Figure 4. 11. $-C_p$ Graphs of Base and all modified airfoils at $\alpha=15^\circ$

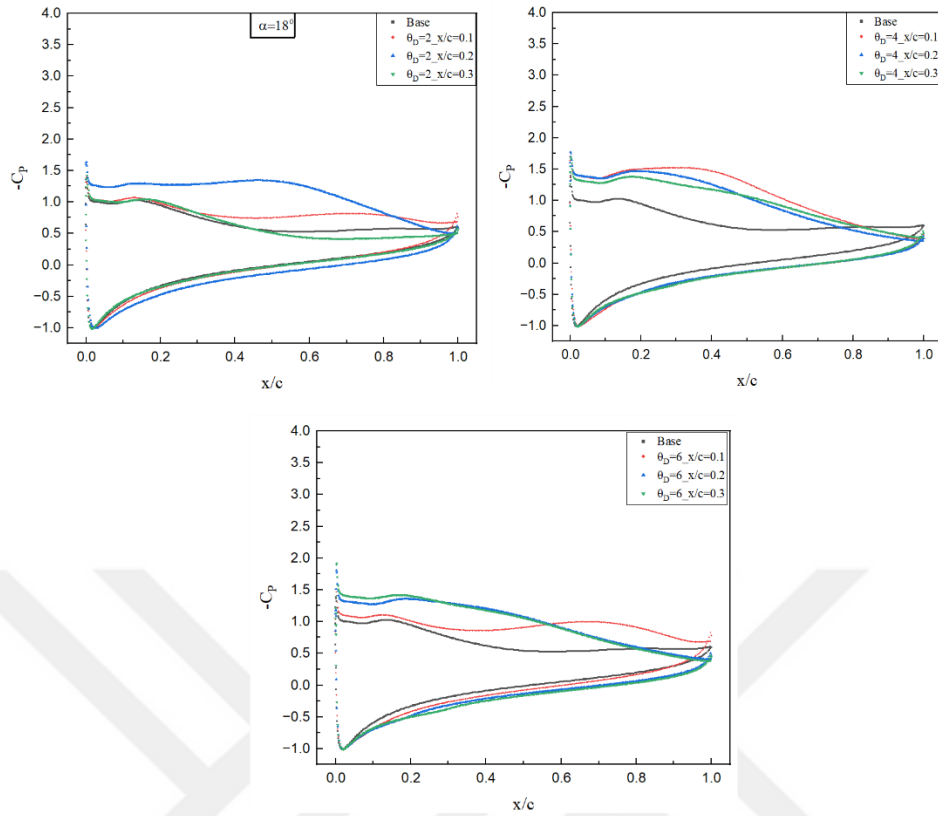


Figure 4. 12. $-C_p$ Graphs of Base and all modified airfoils at $\alpha=18^\circ$

At moderate angles of attack, LSBs are visibly formed, and their positions move from the leading edge to the trailing edge as you move from the base model to the modified models. The models closest to the trailing edge are the models with $x/c=0.3$. From stall angles (15° or 18°), the deflection process increases the aerodynamic performance by increasing the pressure gradient in the pressure part of the airfoils. In addition, at very high angles of attack, the modified models improve the performance by improving the upper curve considerably compared to the base model.

After the $-C_p$ distributions, to better observe the velocity fields around each airfoil, dimensionless velocity contour figures were presented. Figure 4.13 shows the dimensionless velocity contours of the base and modified models with $\nabla=2^\circ$ deflection at low angles of attack ($\alpha=0^\circ, 3^\circ$ and 7°). The same legend was defined for each scenario of each case. According to the results obtained, the velocity values for all cases vary between -0.8 and 1.8. The stagnation point occurs around the leading edge at $\alpha=0^\circ$. From base model to the $\theta_D=2^\circ, x/c=0.3$ model, it

is observed that flow separations shift from the pressure side to the suction side, yielding aerodynamically desirable effects due to the pressure difference between the two surfaces of the airfoil. Furthermore, for the modified models, it is seen that the oscillations in the trailing edge are largely eliminated. This is an effective factor in reducing the drag of the airfoils. In general, all airfoils have a similar profile at $\alpha=3^\circ$. The stagnation point has shifted to the pressure side due to the increasing angle of attack. A high velocity area was formed in the part where the camber angle was the highest. Small separations occurred towards the trailing edge and reduced the local velocity values. When it comes to the angle of attack of 7° , the stagnation point has shifted lower due to the increasing angle. However, in models where the effect of deflection is longer in chord, the high velocity intensity areas have extended towards the trailing edge. In addition, there are LSBs with a stable characteristic. Their thickness is almost as thin as the boundary layer. Since reverse flow also occurs inside the LSB, the color scale here shows that there is a low velocity magnitudes' region. Hence, the position of the LSBs moves towards the trailing edge for the modified models.

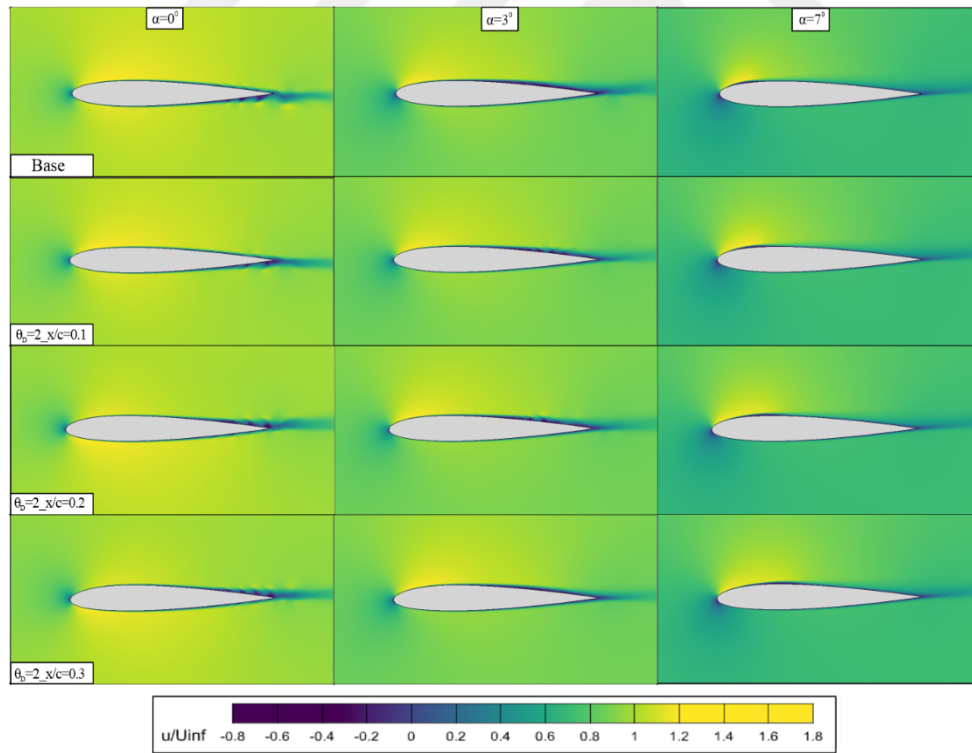


Figure 4. 13. Nondimensional Velocity contours of Base and 2° deflected modified airfoils at low angles of attack

Figure 4.14 shows the dimensionless velocity contours of the base and modified models with a deflection of $\nabla \uparrow 2^\circ$ at high angles of attack ($\alpha = 10^\circ, 15^\circ$ and 18°). At these angles of attack, the flow approaches the airfoil at oblique angles, so the values in the velocity fields around the airfoils are generally lower than at low angles of attack. At $\alpha = 10^\circ$, almost all airfoils have similar velocity contours and the position of the LSBs is not changing. Low velocity fields are available due to the separation around the trailing edge. At $\alpha = 15^\circ$, a large low velocity field appears at the top of the suction section due to beginning of the stall. The higher velocity contours profoundly approach the airfoil surface from base model to the $\theta_D = 2_x/c = 0.3$ model. Also, as in $\alpha = 15^\circ$, the lower velocity contour area becomes smaller. Only in the $\theta_D = 2_x/c = 0.2$ model, unlike the other models, a higher velocity field is observed around the turbulent region. At $\alpha = 18^\circ$, it is seen that the turbulent region in the base model spreads over the entire suction section and becomes quite large. Figure 4.15 shows the dimensionless velocity contours of the base and the modified models with $\nabla \uparrow 4^\circ$ deflection at low angles of attack. Because of deflection at $\alpha = 0^\circ$, the stagnation point in the base model is located at the nose of the airfoil, while in the modified models it is shifted to the regions where the camber is high. Until $\theta_D = 4_x/c = 0.2$ model, vortices occur in the suction side around the trailing edge, while they started to occur in the pressure section from this model onwards. At $\alpha = 3^\circ$, the models similarly have a high velocity zone in the regions where the camber is high, while there are low velocity areas due to flow separation along the chord from about $x/c = 0.4$ distance. At $\alpha = 7^\circ$, there are high velocity areas in the regions where the camber is highest, while the LSBs below these areas shift to the trailing edge with increasing x/c values. As in the 2° deflected models, both the high velocity areas and the positions of the LSBs have shifted towards the rear of the airfoil at this angle of attack.

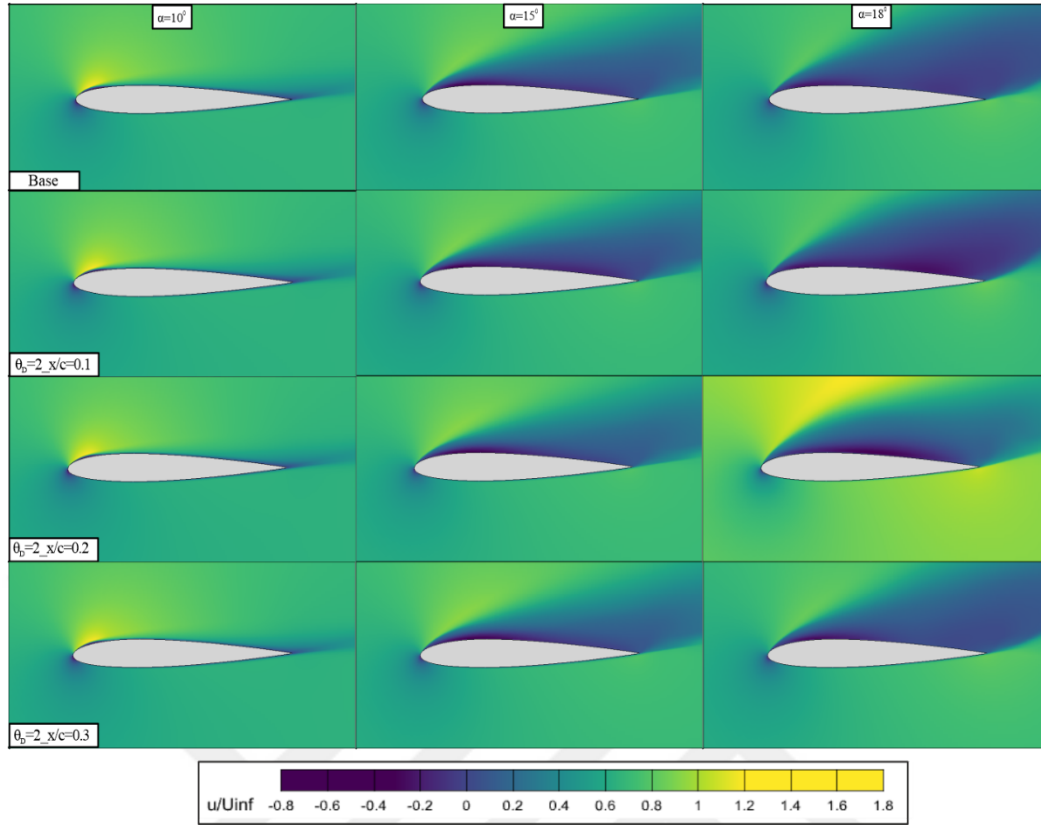


Figure 4. 14. Nondimensional Velocity contours of Base and 2° deflected modified airfoils at high angles of attack

Figure 4.16 shows the dimensionless velocity contours of the base and modified models with 4° deflection at high angles of attack. At $\alpha=10^\circ$, there is again LSB formation, but unlike some previous cases, there is almost no change in the position of the LSB for all cases. At $\alpha=15^\circ$, there is still a low velocity area at the top of the airfoil. However, compared to the base model, this turbulent area of the model with $\theta_D=4_x/c=0.3$ model is greatly reduced. Moreover, in this case, the low velocity area at the top of $\theta_D=4_x/c=0.3$ model is smaller than the low velocity area at the top of the $\theta_D=2_x/c=0.3$ model. Therefore, increasing the deflection angle can be considered as an effective parameter in reducing the low velocity area in the upper region. At $\alpha=18^\circ$, a break occurs starting from $\theta_D=4_x/c=0.1$ and the turbulence field starts to narrow towards the airfoil surface. In the meantime, there is a certain increase in the flow velocity at the boundaries. The smaller LSB may have attracted the flow to itself more and caused the velocity to increase in some local areas. The negative velocity magnitude on the suction side

surface decreased as we moved from the base model to the modified models. This change may indicate that the LSB became smaller and then became more stable.

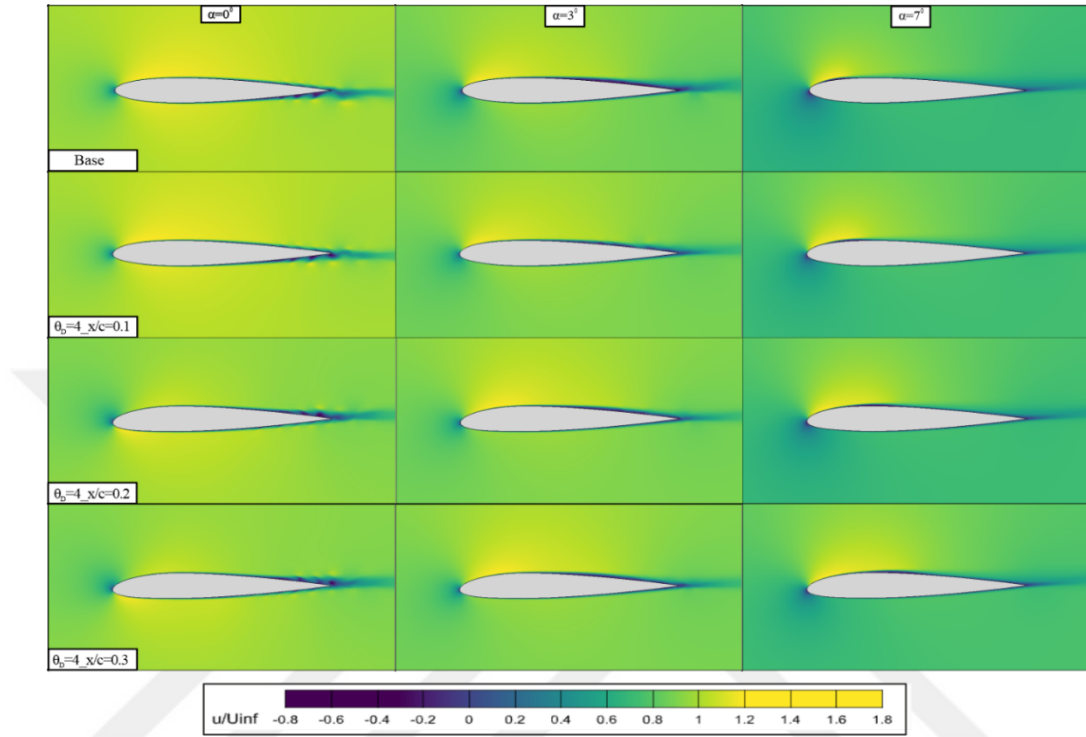


Figure 4. 15. Nondimensional Velocity contours of Base and 4° deflected modified airfoils at low angles of attack

Figure 4.17 shows the dimensionless velocity contours of the base and modified models with 6° deflection at low angles of attack. The vortex layer across the pressure side near the trailing edge of the base model at $\alpha=0^\circ$ became neutral for the $\theta_D=6_x/c=0.2$ model and moved to the suction side for the $\theta_D=6_x/c=0.3$ model. At $\alpha=3^\circ$, the separation layer became thinner from base model to the modified models. At $\alpha=7^\circ$, the high velocity region in the cambered regions is effective for a larger amount of the chord length for the $\theta_D=6_x/c=0.3$ model. The position of the LSB also approaches to the trailing edge, but except for the $\theta_D=6_x/c=0.1$ model.

Figure 4.18 shows the dimensionless velocity contours of the base and modified models with 6° deflections at high angles of attack. At $\alpha=10^\circ$, in contrast to Figure 4.14, the position of the

LSB approaches from the leading edge to the trailing edge and the area of influence of the high velocity field increases on the chord. At $\alpha=15^\circ$, the velocity in the region between laminar flow and turbulent flow increased and the turbulent region narrowed in the modified models compared to the base model.

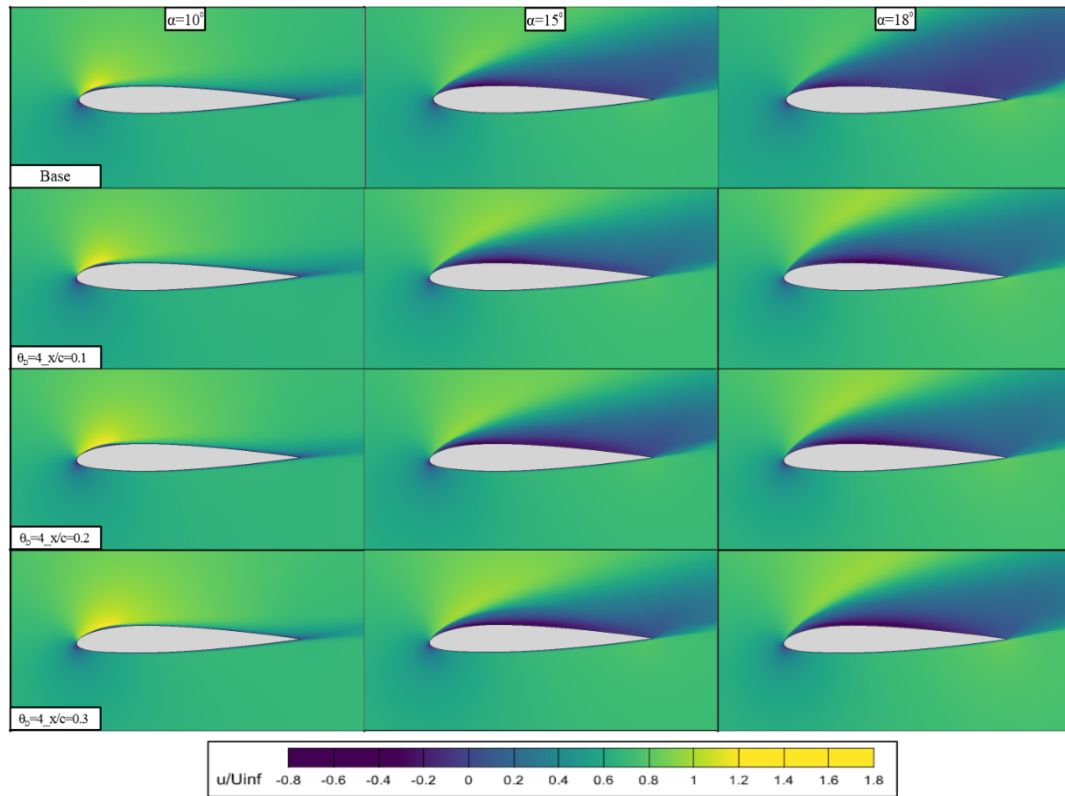


Figure 4. 16. Nondimensional Velocity contours of Base and 4° deflected modified airfoils at high angles of attack

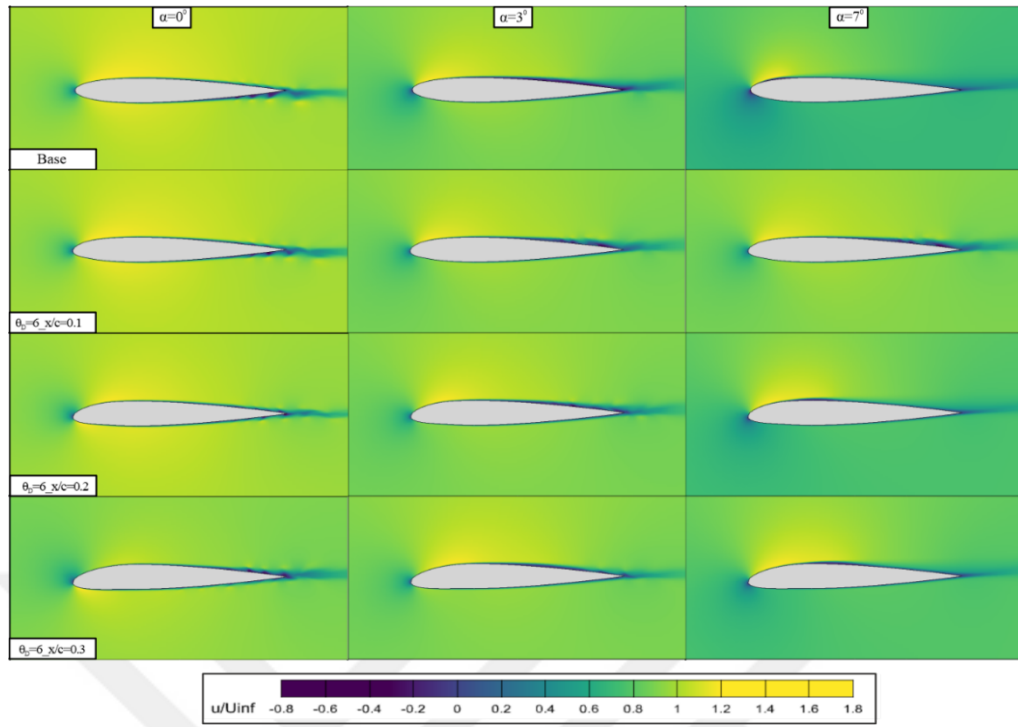


Figure 4. 17. Nondimensional Velocity contours of Base and 6° deflected modified airfoils at low angles of attack

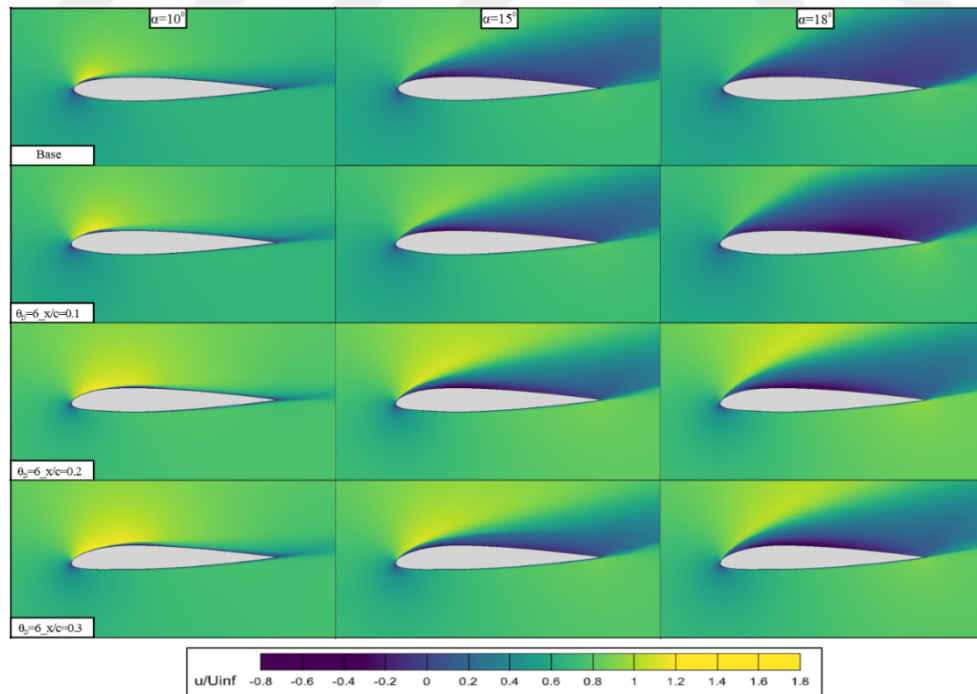


Figure 4. 18. Nondimensional Velocity contours of Base and 6° deflected modified airfoils at high angles of attack

The turbulence kinetic energy (TKE) contours provide the energy distribution of the eddies in turbulent flow that was generated to observe the turbulent flow characteristics around each airfoil. Figure 4.19 shows the TKE contour plots of the base and $\nabla=2^\circ$ deflection models at low angles of attack ($\alpha=0^\circ, 3^\circ$, and 7°). At $\alpha=0^\circ$, the turbulent layer leaving the trailing edge of the base model makes a short oscillation similar to von Karman Street. At $\theta_D=2_x/c=0.2$ model, turbulence energy contours have moved to the upper part of the airfoil because unsteady flow regime is started to spread through downstream region. For the $\theta_D=2_x/c=0.3$ model, this unsteady flow becomes narrower and more compact. While there is a continuous separation from the half chord of the base model at $\alpha=3^\circ$, separation is formed by the dense vorticity eddies to the surrounding area at certain intervals for the modified models. At $\alpha=3^\circ$, it can be observed that TKE intensity of modified airfoils were reduced compared to base case. The effect of modified geometries can be seen at this angle of attack as a flow control element specifically for $\theta_D=2_x/c=0.3$ model. At $\alpha=7^\circ$, there is a TKE contours focused on the pressure side of the airfoils for each case close to the leading edge, which is most likely caused by the LSB. From the base model to the $\theta_D=2_x/c=0.3$ model, the LSB is closer to the trailing edge, but the presence of a turbulent region up to the downstream region.

Figure 4.20 shows the TKE contour plots of the base and 2° deflected models at high angles of attack ($\alpha=10^\circ, 15^\circ, 18^\circ$). At $\alpha=10^\circ$, low intensity of TKE contours can be seen near to trailing edge of the airfoils for all cases as expected for a pre-stall condition. Also, similar flow characteristics were interpreted from the aerodynamic performance of the airfoils seen in the lift coefficient magnitudes for this specific angle of attack. At $\alpha=15^\circ$, due to the stall, TKE contours intensity at the trailing edge for base case, while these profound contours move to leading edge with the increase at the initiation positions of deflection angles (i.e., from $0.1c$ to $0.3c$). Especially from $\theta_D=2_x/c=0.2$ model, this intense turbulent region approaches the airfoil surface and is concentrated here. In addition, the intensity of the TKE from the trailing edge and formed in the downstream region decreases from the base model to $\theta_D=2_x/c=0.3$ model. At $\alpha=18^\circ$, in addition to TKE intense contours through downstream of the all models, it is observed that higher turbulence magnitudes at the pressure side (i.e., upper part) of $\theta_D=2_x/c=0.2$ model.

Figure 4.21 shows the TKE contour plots of the base and $\theta=4^\circ$ deflected models at low angles of attack. At $\alpha=0^\circ$, there is still oscillatory flows towards the trailing edge for all models. With the increase of angle of attack, the TKE contours were in a smoother shape specifically for the modified geometries. Also, with the comparison of $\theta=2^\circ$ in Figure 4.19, the increase of deflection angle enhances the flow control results, emphasizing the TKE intensity at $\alpha=3^\circ$. At $\alpha=7^\circ$ and $\alpha=10^\circ$, pre-stall conditions can be easily observed. LSB location moves from leading edge to trailing edge with the increase of initiation position of deflected surface. Especially in the case of $\theta_D=4_x/c=0.3$, the LSB is shifted towards the rear of the airfoil. Therefore, the induced drag due to lift is less produced in this modified model relative to the base case.

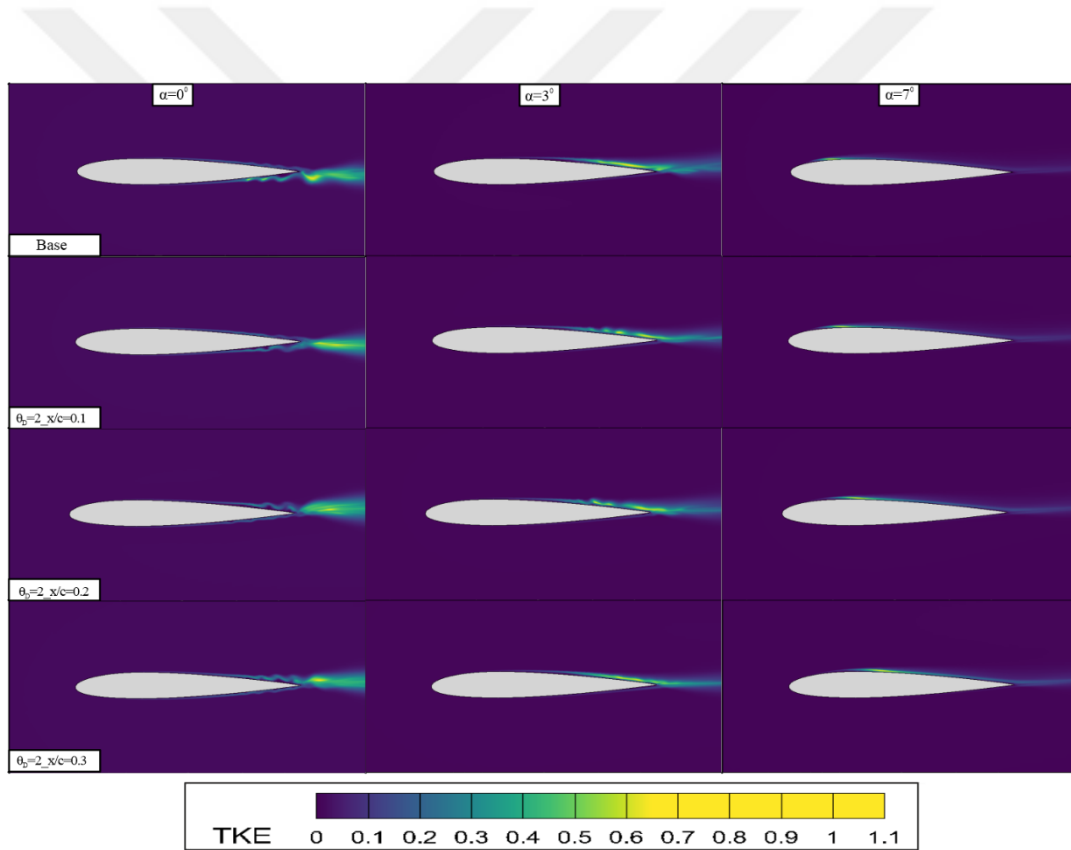


Figure 4. 19. TKE contours of Base and 2° deflected modified airfoils at low angles of attack

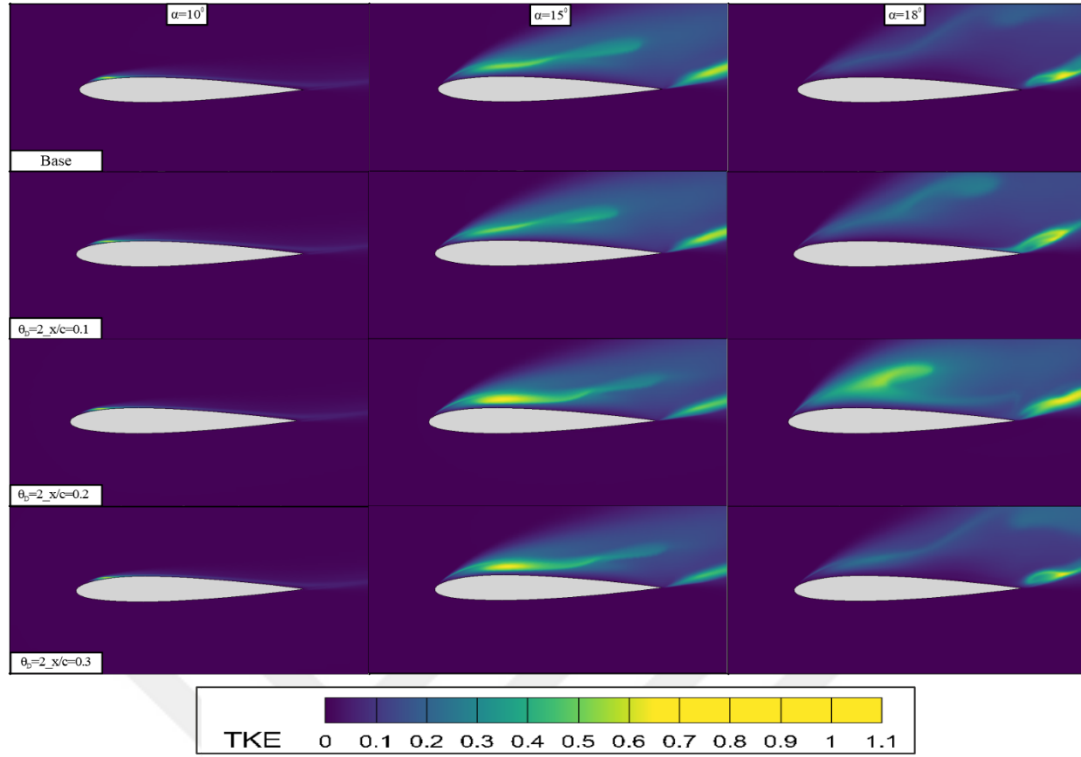


Figure 4. 20. TKE contours of Base and 2° deflected modified airfoils at high angles of attack

Figure 4.22 shows the TKE contour plots of the base and $\theta=4^\circ$ deflected models at high angles of attack. After pre-stall condition at $\alpha=10^\circ$, the intense TKE contours can be seen at $\alpha=15^\circ$. They profoundly shifted from trailing edge to leading edge with the increase of initiation position of the deflected surface. A similar trend was also observed at $\alpha=18^\circ$. Therefore, high lift is attained due to the modified airfoils of $\theta_D=4_x/c=0.3$ model.

Figure 4.23 shows the TKE contour plots of the base and $\theta=6^\circ$ deflection models at low angles of attack. For $\alpha=0^\circ$, the effect of the modification on the flow structure around the airfoil appears to be stable. However, unlike the $\theta=4^\circ$ deflected models, the $\theta=6^\circ$ deflected configurations exhibit strong flow separation on both the upper and lower surfaces. This double-sided separation is likely due to the increased curvature on the suction side, which significantly decelerates the flow, preventing it from remaining attached to the surface. From $\alpha=3^\circ$ to $\alpha=7^\circ$, the intensity of TKE contours shifts from the leading edge toward the trailing edge for all configurations. Moreover, the upstream initiation position of deflection in the modified airfoils enhances their flow control capabilities. Especially, the turbulent region in the wake of the

$\theta_D=6_x/c=0.3$ model is significantly smaller than that of the base case. The effectiveness of these models in reducing wake turbulence is even greater than that observed for the $\theta=4^\circ$ deflected configurations. Therefore, at this angle of attack, the $\theta=6^\circ$ deflected models demonstrate the highest drag reduction performance.

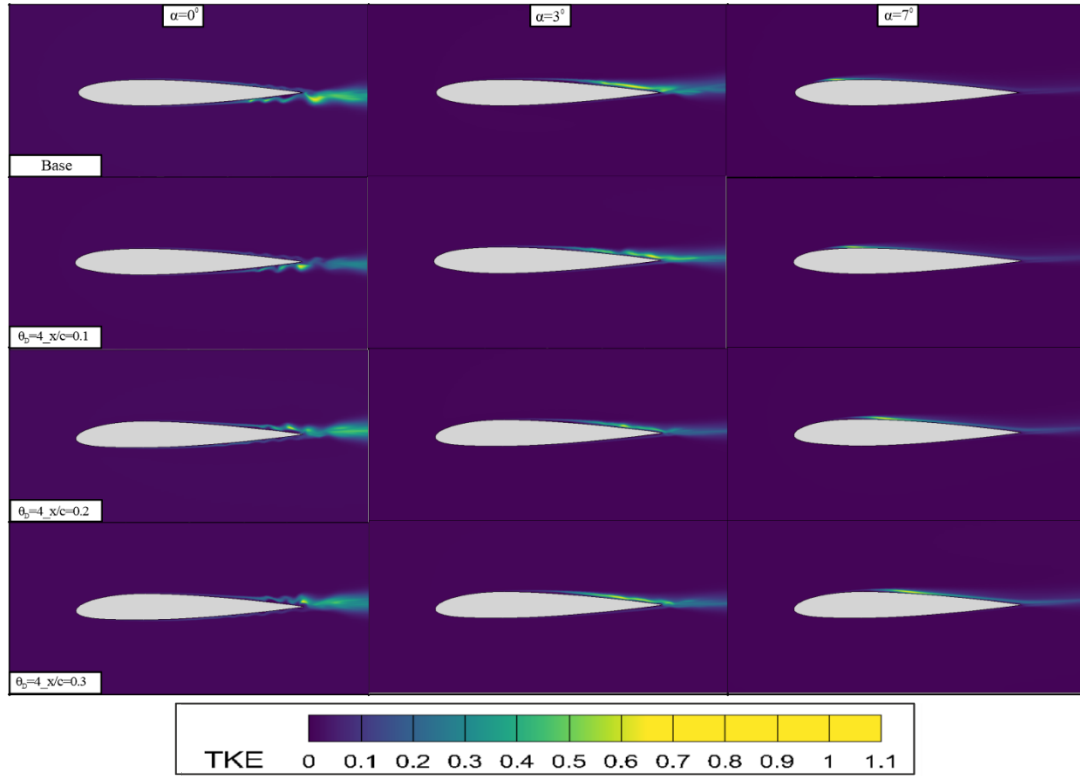


Figure 4. 21. TKE contours of Base and 4° deflected modified airfoils at low angles of attack

Figure 4.24 shows the TKE contour plots of the base and $\theta=6^\circ$ deflected models at high angles of attack. From $\alpha=10^\circ$ to $\alpha=15^\circ$, the most notable case is seen for $\theta_D=6_x/c=0.3$ model. By the effect of control element, TKE intensity is lower than the other models. Additionally, the $\theta_D=6_x/c=0.1$ and $\theta_D=6_x/c=0.2$ models decrease the TKE magnitudes at the trailing edge compared to the base case. A similar trend was also observed for $\alpha=18^\circ$. Specifically, the turbulence distribution of the $\theta_D=6_x/c=0.3$ model and the turbulence model of the $\theta_D=4_x/c=0.3$ model is almost identical, but the intensity of the high turbulence areas in the $\theta_D=6_x/c=0.3$ model has decreased.

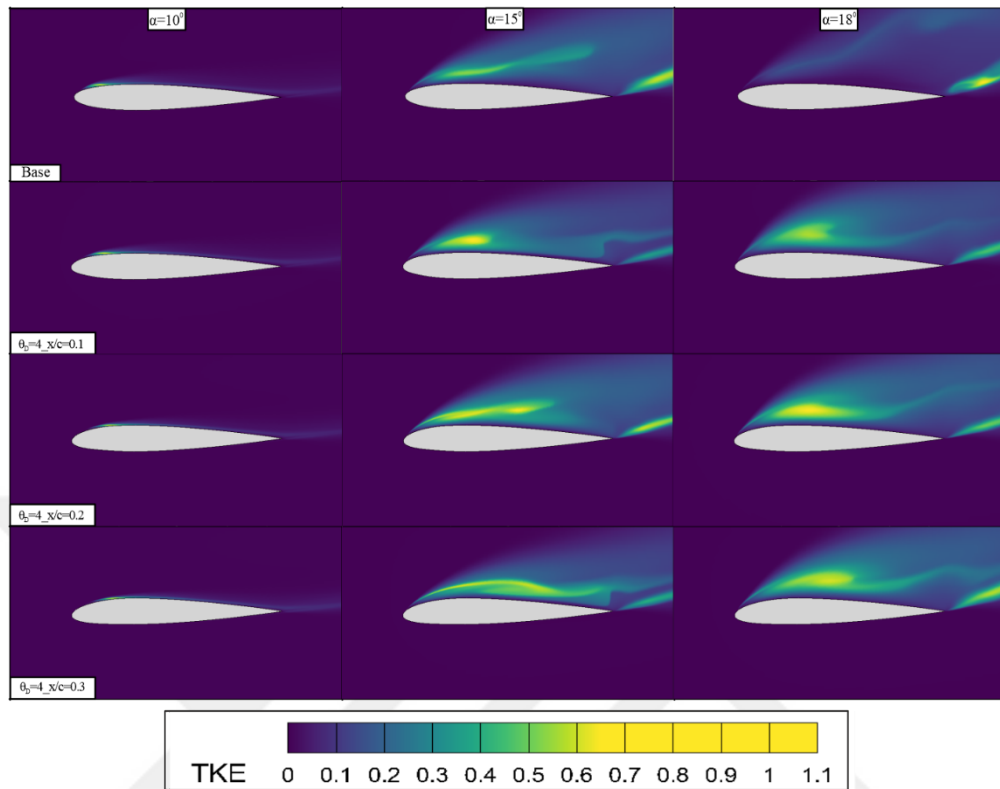


Figure 4. 22. TKE contours of Base and 4° deflected modified airfoils at high angles of attack

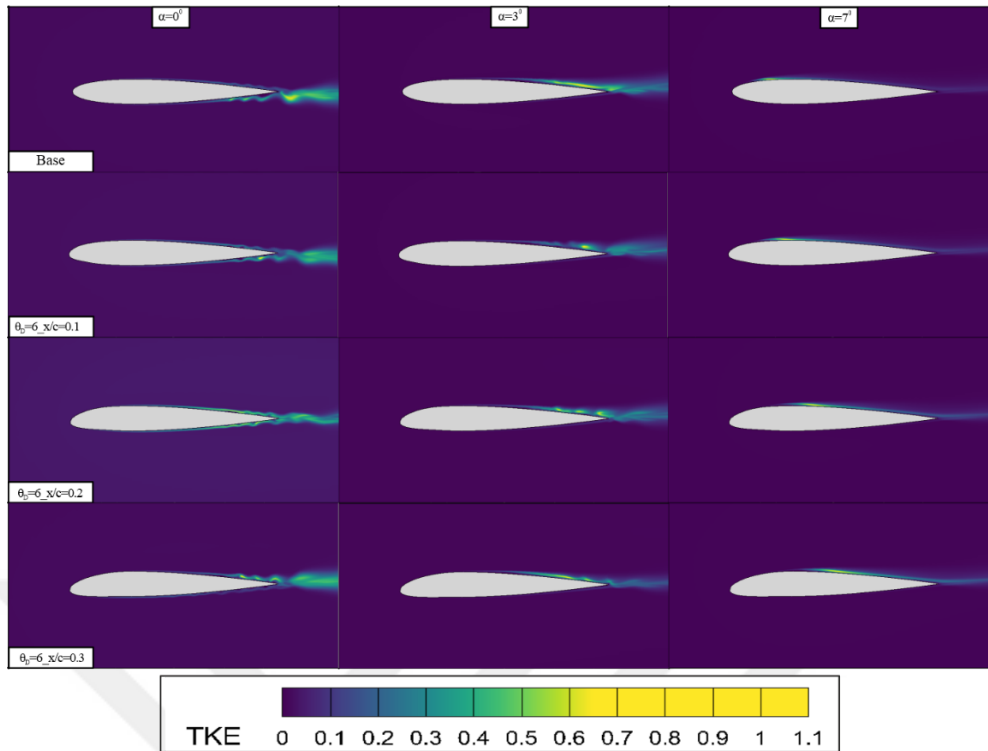


Figure 4. 23. TKE contours of Base and 6° deflected modified airfoils at low angles of attack

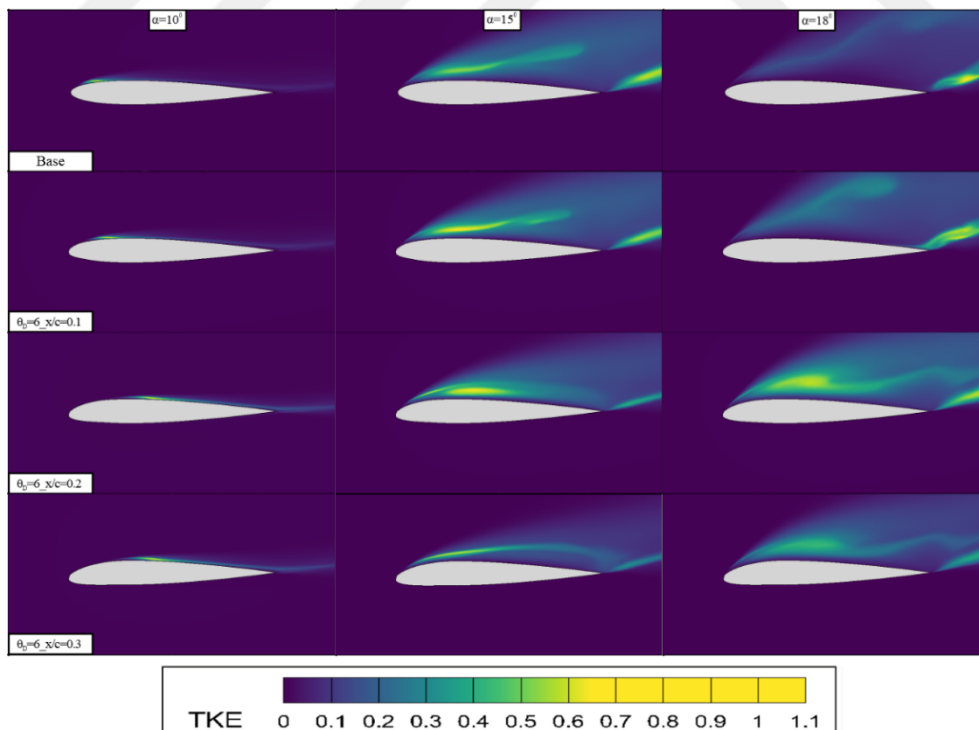


Figure 4. 24. TKE contours of Base and 6° deflected modified airfoils at high angles of attack

Moreover, streamline topology plots were generated to provide a clearer understanding of the flow structures and to identify flow separation regions more effectively. For this purpose, both the base and modified models were compared at low angles of attack ($\alpha = 0^\circ, 3^\circ$, and 7°), followed by a comparison at higher angles of attack ($\alpha = 10^\circ, 15^\circ$, and 18°). Figure 4.25 presents the streamline topology of the base model and the $\theta = 2^\circ$ deflected models at low angles of attack. For $\alpha = 0^\circ$, flow separation occurs near the trailing edge for all configurations, and no distinct focal region is observed. At $\alpha = 3^\circ$, separation on the upper side leads to the formation of unstable vortices. For instance, in the embedded figure for the $\theta_D = 2^\circ, x/c = 0.2$ model, a focal point is clearly visible on the upper side of the airfoil. At $\alpha = 7^\circ$, a LSB emerges in all models, shifting progressively from the leading edge toward the trailing edge.

Figure 4.26 displays the streamline topology for the base and $\theta = 2^\circ$ deflected models at higher angles of attack. For $\alpha = 10^\circ$, the location of the LSB remains nearly unchanged across all models, consistent with trends observed in the TKE distributions. At $\alpha = 15^\circ$, two focal regions appear on the upper surface of the base model: one near the leading edge and another near the trailing edge. As the initiation position of the deflected models increases, one of these foci becomes dominant and shifts toward the leading edge. At $\alpha = 18^\circ$, as in the previous angle of attack, two recirculation zones are observed on the upper surface, except for a small focal point at the trailing edge across all configurations. For $\theta_D = 2^\circ, x/c = 0.1$ and $\theta_D = 2^\circ, x/c = 0.2$ models, only a single primary vortex layer forms. In contrast, $\theta_D = 2^\circ, x/c = 0.3$ model, exhibits both primary and secondary vortex layers, although these are significantly smaller than those in the other configurations. Additionally, a weaker vortex layer detaches from the trailing edge; its intensity increases from the base model to the $\theta_D = 2^\circ, x/c = 0.1$ model and then diminishes thereafter.

Figure 4.27 exhibits the streamline topology plot of the base and $\theta = 4^\circ$ deflected models at low angles of attack. At $\alpha = 0^\circ$, as in the $\theta = 2^\circ$ deflected models, flow separation is seen toward the trailing edge for all cases, and no foci is available. At $\alpha = 3^\circ$, the foci emerged from the top surface of the airfoil to form vorticities as seen in the embedded figures. When the angle of attack increased to 7° , the curling vortex layer started to form LSB.

Figure 4.28 shows the streamline topology plot of the base and $\theta = 4^\circ$ deflected models at high angles of attack. At $\alpha = 10^\circ$, the position of the LSBs changed slightly compared to the base model. At $\alpha = 15^\circ$, at this post-stall condition, only the primary vortex was formed in all models

except the base model. The center of the vortex foci moved closer to the trailing edge with an increasing x/c value. In this deflection model, LSB can get closer to the trailing edge at higher angles of attack. At $\alpha=18^\circ$, the base model had two recirculation zones on the upper surface whereas all the modified models had a single vorticity region, and its thickness decreased with increasing x/c value. Also, the trailing edge vortex, that is available for the base case, was eliminated in all modified models. In addition, although the vortex centers of the $\theta_D=4^\circ$ $x/c=0.1$ and $\theta_D=4^\circ$ $x/c=0.2$ models are almost at the same place, the center point of the $\theta_D=4^\circ$ $x/c=0.3$ model is closer to the trailing edge. This shows us one of the most important effects of the x/c parameter.

Figure 4.29 shows the streamline topology graph of the base and $\theta=6^\circ$ deflected models at low angles of attack. At $\alpha=0^\circ$, flow oscillations at the trailing edge can be observed for all cases, whereas modified airfoils have less impact on the downstream. At $\alpha=3^\circ$, small vortices are formed from the flow separating towards the trailing edge due to the increasing momentum and energy of the flow. At $\alpha=7^\circ$, the LSB formed on the upper surface of the airfoils approaches the trailing edge with increasing x/c value. This feature is observed very limited in the models with 2° deflection. In this sense, the models with the ability to delay the LSB at this angle of attack are the models with 4° and 6° deflection.

Figure 4.30 shows the streamline topology graph of the base and $\theta=6^\circ$ deflected models at high angles of attack. At $\alpha=10^\circ$, the LSB moves closer to the trailing edge with increasing x/c value. This is a feature of the 6° deflected models to reveal the flow control phenomena for pre-stall conditions. It should be noted that it may not always be desirable for the LSB to offset towards the trailing edge. In the lift distributions of the airfoils, this statement is generally valid as the most lift is usually produced near the leading edge, and in order to analyze the lift obtained correctly, it is necessary to obtain the pressure distributions along the chord of each model. At $\alpha=15^\circ$, a flattened LSB can be seen starting, hence it can be said that the lowest drag occurs for the $\theta_D=6^\circ$ $x/c=0.3$ model. At $\alpha=18^\circ$, there is also a layer that has separated from the trailing edge and curled to form a vortex. This layer is eliminated from $\theta_D=6^\circ$ $x/c=0.2$ model and the turbulent field continues to become smaller and more compact at the $\theta_D=6^\circ$ $x/c=0.3$ model.

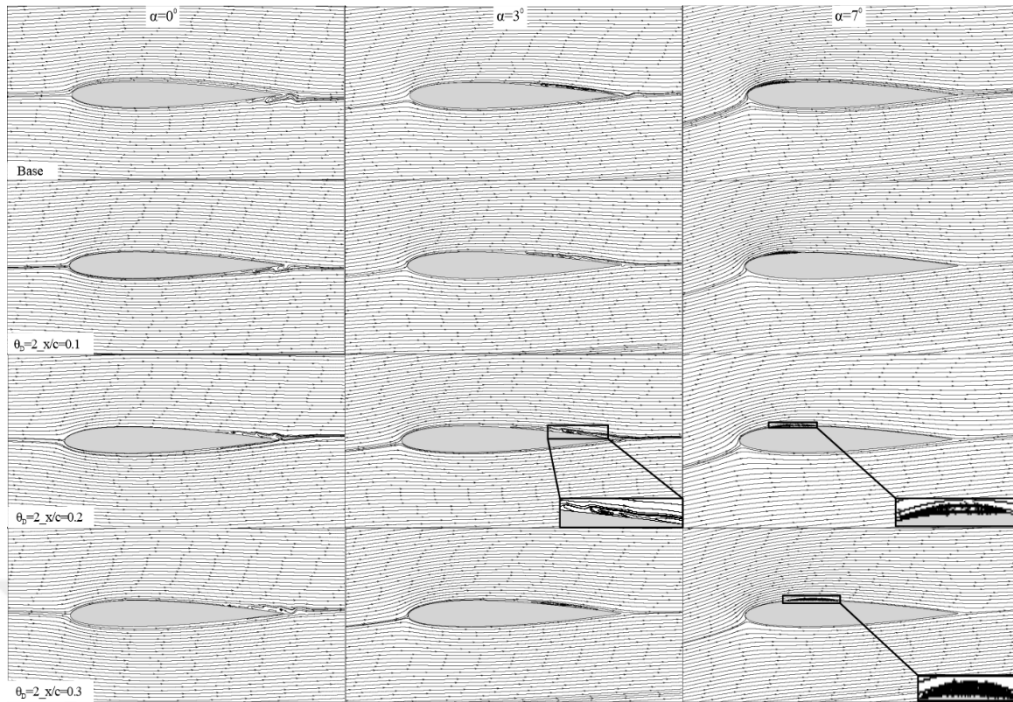


Figure 4. 25. Streamline Topology of Base and 2° deflected models at low angles of attack

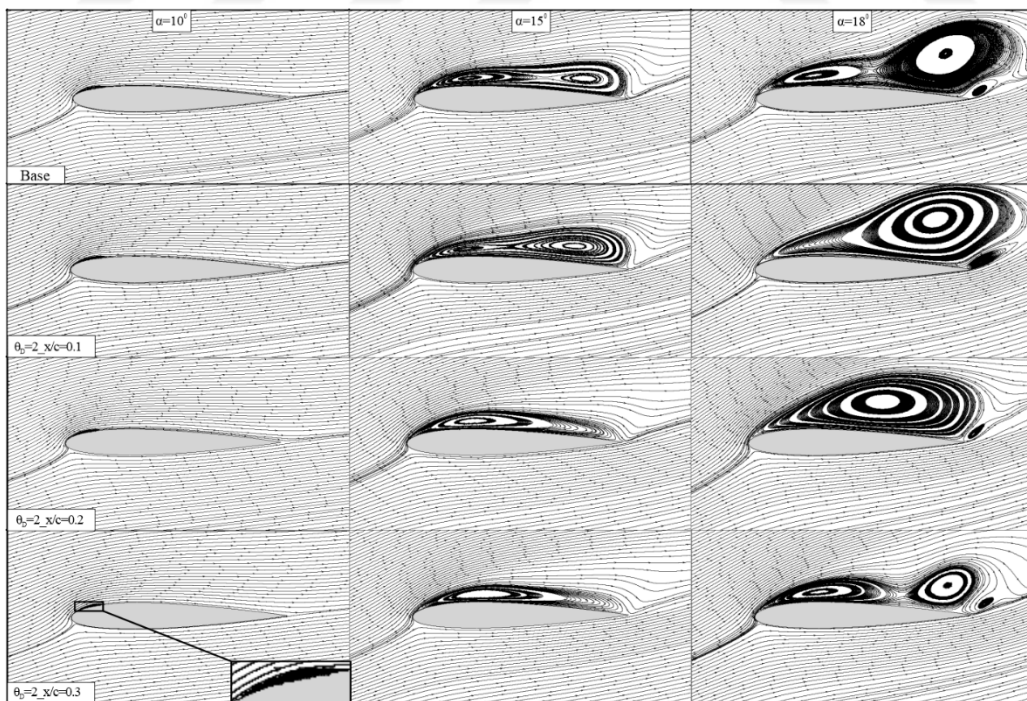


Figure 4. 26. Streamline Topology of Base and 2° deflected models at high angles of attack

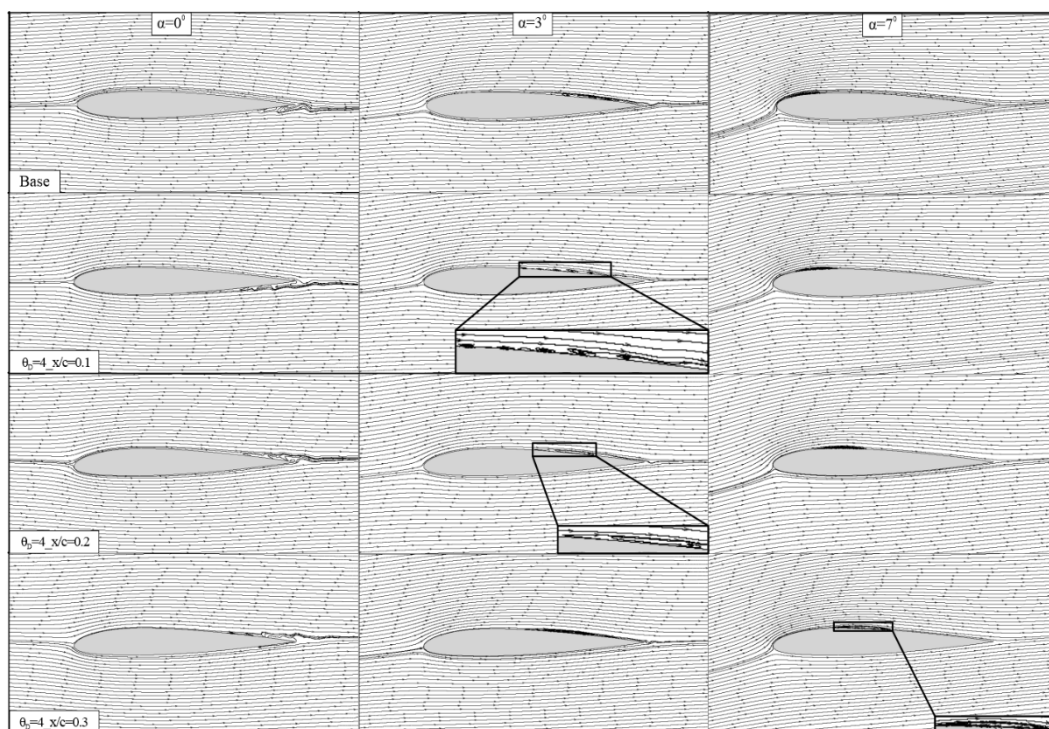


Figure 4. 27. Streamline Topology of Base and 4° deflected models at low angles of attack

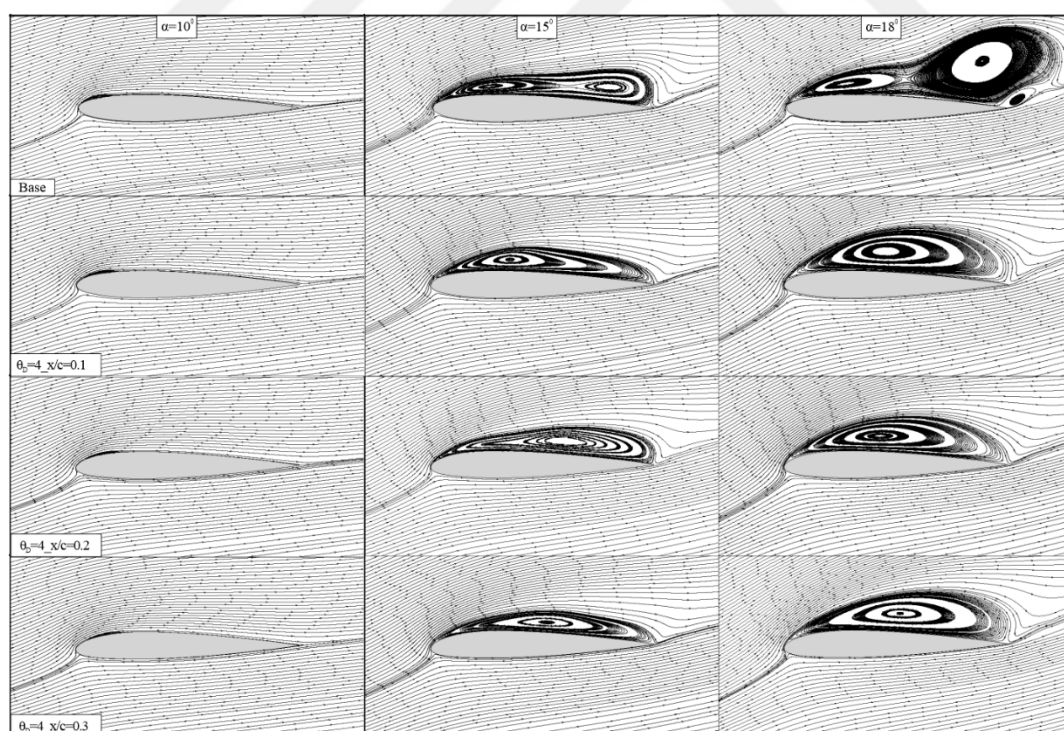


Figure 4. 28. Streamline Topology of Base and 4° deflected models at high angles of attack

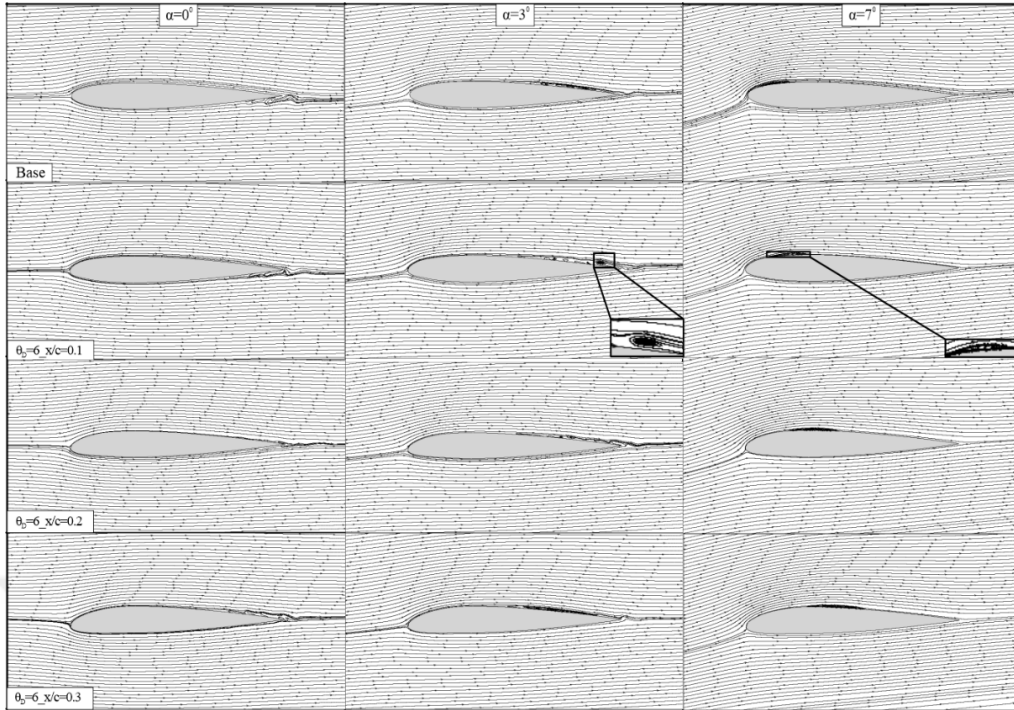


Figure 4. 29. Streamline Topology of Base and 6° deflected models at low angles of attack

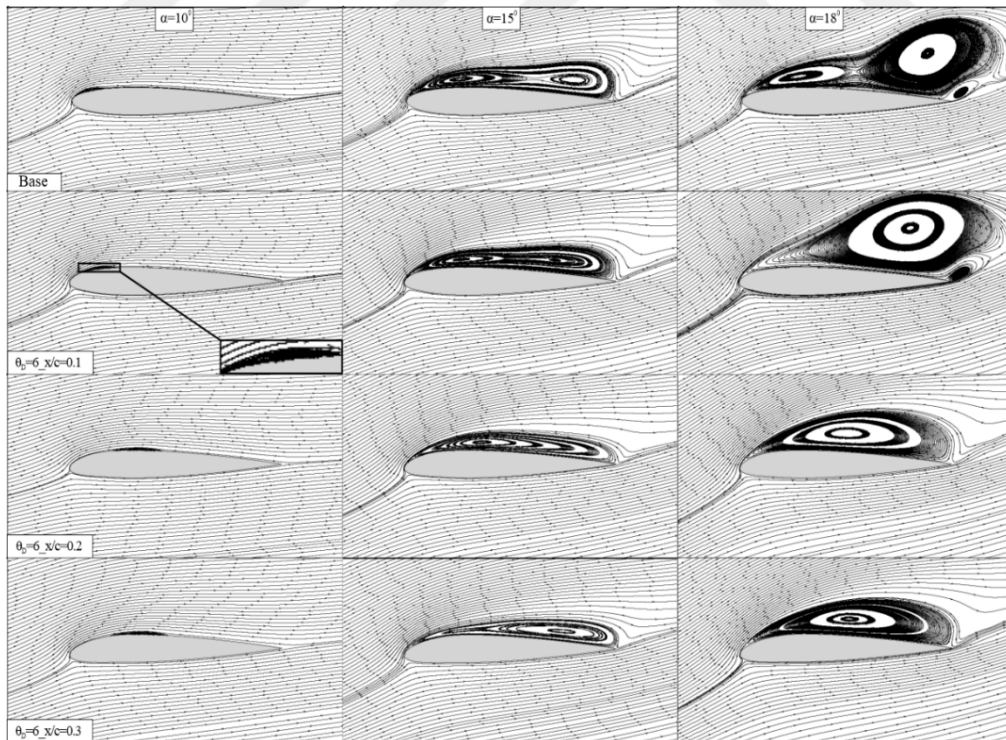


Figure 4. 30. Streamline Topology of Base and 6° deflected models at high angles of attack

5. CONCLUSIONS

In this thesis, the aerodynamic performance of a NACA 0012 airfoil with a morphing leading edge was numerically investigated at a Reynolds number of $Re=1 \times 10^5$. The modified configurations were created by introducing leading-edge deflections of 2° , 4° , and 6° , applied at varying chordwise positions ($x/c = 0.1, 0.2$, and 0.3). Comparative analyses between the base airfoil (non-deflected model) and the modified geometries were conducted at multiple angles of attack ($\alpha = 0^\circ, 3^\circ, 7^\circ, 10^\circ, 15^\circ$, and 18°) using ANSYS FLUENT. Streamline visualizations and contour plots were employed to assess the impact of morphing on flow features such as laminar separation bubbles (LSB), vortex shedding, and turbulence intensity.

The results indicate that the 4° and 6° deflection models generally outperformed the 2° deflection configuration in terms of aerodynamic efficiency across the range of angles of attack. Particularly beyond $\alpha = 10^\circ$, the baseline model exhibited a rapid increase in drag, whereas the modified models maintained more favorable drag characteristics. Among all configurations, the $\theta_D=4^\circ$ $x/c=0.3$ model showed the most significant performance gain at $\alpha=10^\circ$, achieving a 37.75% reduction in drag and a 73.61% increase in lift-to-drag ratio (C_L/C_D) compared to the base case. At higher angles of attack, especially in the post-stall regime (e.g., $\alpha = 18^\circ$), the 6° deflection models proved more effective, exhibiting improved flow attachment and suppression of trailing-edge vortex structures. Streamline analyses also revealed that, at moderate angles such as $\alpha = 7^\circ$ and 10° , the modified models helped shift the LSBs closer to the trailing edge, although only the 6° deflection models were effective at $\alpha = 10^\circ$.

While the morphing approach demonstrated significant gains in aerodynamic efficiency and flow control, it did not significantly delay the onset of stall compared to the base airfoil. Nevertheless, the reduction in turbulence and flow separation in the stall regime suggests that leading-edge deflection may offer considerable advantages under certain operating conditions. Future studies may explore higher angles of attack to better characterize post-stall behavior, as well as investigate the performance of leading-edge morphing on asymmetric airfoils. Experimental validation and unsteady simulations could further enhance the understanding of the aerodynamic effects and practical applicability of this morphing approach.

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