

**COMPUTATIONAL AND EXPERIMENTAL INVESTIGATION  
OF AERODYNAMIC PARAMETERS ON DYNAMIC CAMBER  
AEROFOIL FOR SUBSONIC RANGE**

Thesis Submitted For the Award of the Degree of

**DOCTOR OF PHILOSOPHY**

**in**

**Aerospace Engineering**

**By**

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**LOVELY PROFESSIONAL UNIVERSITY  
PUNJAB  
2022**

## **Declaration**

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The work presented in this dissertation was carried out at the School of Mechanical Engineering at Aerospace Department Lovely Professional University between August 2018 to November 2021 under the supervision of Dr J V Muruga Lal Jeyan.

I hereby declare that, except for where specific references are made to the work of other authors or specially indicated in the text, the contents of this dissertation are his work. This report has not been submitted in parts or whole to any other university for a degree, diploma, or other qualification. The work is purely done by me and includes nothing, which is the outcome of work done in collaboration.

A handwritten signature in blue ink, appearing to be 'Bilji C M', on a light yellow background.

Bilji C M

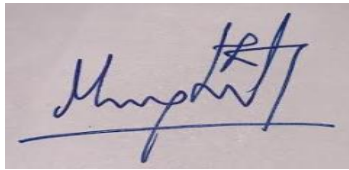
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## Certificate

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This is to certify that **Mr Bilji C M** has completed the PhD. Under my guidance and supervision, aerospace engineering was titled "**Computational and Experimental Investigation of Aerodynamic Parameters on Dynamic Camber Aerofoil for Subsonic Range**". To the best of my knowledge, the present work results from his original investigation and study. No part of this thesis work has ever been submitted for any degree.

This thesis is fit for the submission for the partial fulfilment of the condition for the award of the degree of PhD in Aerospace Engineering.

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---

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Bilji C M

## **Abbreviations and symbols**

|                      |                                             |
|----------------------|---------------------------------------------|
| NACA.....            | National Advisory committee for Aeronautics |
| DARPA.....           | Defense Advanced Research Projects Agency   |
| TDT.....             | Transonic Dynamic Tunnel                    |
| CFD.....             | Computational Fluid Dynamics                |
| FEM.....             | Finite Element Method                       |
| AoA.....             | Angle of Attack                             |
| UAV.....             | Unmanned Aerial Vehicle                     |
| MAV.....             | Micro Aerial Vehicle                        |
| LE/TE.....           | Leading-edge/Trailing edge                  |
| AAW.....             | Active Aero elastic wing                    |
| Cl.....              | Coefficient of Lift                         |
| Cd.....              | Coefficient of drag                         |
| S <sub>f</sub> ..... | Stagnation front                            |
| S <sub>a</sub> ..... | Stagnation aft                              |

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## **Abstract**

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Morphing is a term used in the aerospace industry to describe a group of technologies that improve a vehicle's performance by modifying specific characteristics to better adapt the vehicle's condition to the environment and mission at hand. The wing morphing concept has huge potential and capability as it can increase wing efficiency and remove the limitation in aircraft flight manoeuvres. Researchers are making different morphing wing designs or modifying the existing designs to increase aircraft flight characteristics. The primary purpose of this research is to make fixed-wing highly efficient, as we know that birds morph their wings efficiently under various conditions. Often, histories are the best answer for future problems, so scientists started looking for the mechanisms in the past and then started solving their concerns with available advanced technology. In this dissertation, we aim to highlight the different types of in-plane and out plane morphing techniques developed in the past decades, mechanisms used, material, and the development and the incorporation of these techniques in modern-day aircraft and extend the research on airfoils with varying the camber along the chord line. The primary challenge was to narrow down to a particular NACA series since there are many different NACA series available. An extended review of the articles and research led us to NACA 5 digit series. Further, it narrowed down to NACA 23012 series. Our next step was to design a suitable mechanism to vary the chord length and then manufacture the airfoils with variable camber mechanism testing them in the wind tunnel and Ansys simulation software at different low subsonic speeds and calculating the lift achieved by using the empirical lift formulae and comparing the results observed computationally and experimentally received, At last results, conclusion and future scope is elaborately discussed from the current experimental study on 5 digit normal and variable camber NACA airfoil along with plots.

# CHAPTER I

## INTRODUCTION

# **1 Introduction**

## **1.1 Overview**

Over the years, considerable advances have been achieved in the field of aeronautics. The journey began on December 16 1903, when Wright flyer<sup>1</sup> took off for 30 seconds and created history for the first man-made object to fly, which was heavier than air aircraft class. Wright Brothers were credited for their achievement, But it couldn't have been a success for them without a proper wing design. To achieve the desired results, Wright Brothers extensively relied on data accumulated over the years by aeronautic pioneers and took a leaf out of the data and designed a proper bi-plane wing design which helped them to create history, the quest to fly higher, safer, and efficiently drives engineers to explore new concepts and new wing designs in aircraft to increase stability, range and performance, many advances has been made in the field of wing design. Considerable design changes have been adapted to increase Lift generation.

### **Aerofoil nomenclature**

Different types of aerofoil series have been used in aircraft over the decades. These are designed perfectly to achieve the results during different phases of flight manoeuvres. The National Advisory Committee for Aeronautics in America devised a system where the inscription and statistics denote the airfoil's main features. These are classified as 4digit, 5digit, and 6digit NACA series, where each digit signifies different values of variables which could be the prime factor to induce variation in the lift.

The numerical values which have the most result on the character of an airfoil is mentioned below

- i) the deviation of camber from the centre line
- ii) the location of the highest value of camber in terms of chord length
- iii) the utmost width and deviation of thickness along the chord

NACA 4 and 5 digit series are designed for aeroplane flying subsonic speed range, 6, 7 and NACA, 8 digit series, are designated as high-speed airfoils. In every NACA digit series, there are

complex formulae for predicting the coefficient of lift, camber, the maximum position of camber and maximum thickness of aerofoil etc.; for instance, let us look at the 5 digit system.

- i) The designated coefficient of lift value is calculated from the first digit. The first is three halved to get the coefficient of lift value
- ii) Maximum camber is shown by the first two-digit halved value
- iii) Maximum thickness is calculated by the last two digits

Apart from wing profile shape, different taper ratio and aspect ratio wings were used to increase the coefficient of lift and performance of the aircraft.

### **Lift augmentation devices**

Several experiments have been conducted to present aerofoil's a new variety of dynamic mechanisms which actually change camber during the flight so that it helps a pilot to change his aeroplane from a high lift device to a low-speed subsonic aeroplane at will. The incredible improvement obtained by such a mechanism will bring such an impact on aviation history. Many patents have been applied for and granted. Although there are merits and demerits which will be not fair to be compared when we see the advantages obtained from the augmentation devices, Another important consideration is the simplicity of the device.

### **Lift Augmentation devices (Flaps and Slats/Slots)**

Lift Augmentation devices can be used together or single-handedly by the pilot depending upon the requirement, It's the pilot who decides which combination is required, and correspondingly the command is given. The control surface takes action, thanks to the Fly By Wire system, which makes pilot at ease during these complex manoeuvres.

- i) Fixed Slots
- ii) Controlled Slots
- iii) Automatic Slots
- iv) Blown Slots

Subdivision of Flaps:

- i) Camber Flaps
- ii) Split Flaps
- iii) Slotted Flaps

- iv) Lift Flaps
- v) Blown Flaps
- vi) Jet Flaps
- vii) Nose Flaps
- viii) Spoilers
- ix) Lift Dumpers
- x) Air brakes

There is an increment in performance characteristics of an aerofoil with the implementation of both Slots and Flaps. Below mentioned are the increment in performance characteristics.

- i) Lift is increased
- ii) Drag is increased
- iii) Variation in stalling angle
- iv) Lift reduction at will
- v) Effect of change in trim angle

### **Leading Edge Flaps(Slots)**

If we introduce a tiny auxiliary airfoil in front of the leading edge of the main wing, with an appropriate space or slot in connecting, the lift coefficient of the aerofoil can be increased to 60%, and stalling angle can be increased from  $15^\circ$  to  $22^\circ$  or more, which is an added advantage while considering the problem of landing an aircraft. Due to the breaking of stream line flow on the top of the aerofoil, stalling in flight occurs. With slot arrangement, airflow can be made to flow close to the surface of the airfoil; hence staling can be delayed, and more smooth flow is achieved on top of the airfoil at a higher angle of attack, and literature survey has provided data which instigate that it helps to increase lift value. The increase in lift is helpful while landing and takeoff of an aeroplane, but it becomes a un necessity at high-speed flights, so we prefer an automatic slot system that retracts in and out during the requirement phase.

### Important points regarding Slats and Slots

- i) To maintain smooth flow over the surface, blown slots are used, which blows air on the top of the airfoil to increase the momentum of the retarding air and further move them along through maintaining smooth flow.
- ii) The slot idea can be extended to other aircraft parts. Specially designed fillets and cowlings can be used to smoothen the flow around aircraft to prevent airflow separation.
- iii) The extension of the slot idea can be applied to different mission profiles of an aeroplane during its movement in the air. This idea can be applied to different types of manoeuvres

|                                                                                                     |     |     |                                                                                                        |
|-----------------------------------------------------------------------------------------------------|-----|-----|--------------------------------------------------------------------------------------------------------|
| <br>Slotted wing   | 40% | 20° | Controls boundary layer. Slight extra drag at high speeds.                                             |
| <br>Fixed slat    | 50% | 20° | Controls boundary layer. Extra drag at high speeds. Nose-up pitching moment.                           |
| <br>Movable slat | 60% | 22° | Controls boundary layer. Increases camber and area. Greater angles of attack. Nose-up pitching moment. |

Fig1-1 High lift devices (slats and slots)

### Flaps

The flaps work on the same principle as aileron functions. It deflects downward to create lift due to extra surface area when in need, to provide descent glide angle and reduction in speed during landing and fly with cruising speed with flaps up. With the invention of slots, it was believed flaps would be forgotten, but it was not as taught that they are flying necessity these days in modern aircraft. Flaps like slots can increase lift, and these flaps slots can increase drag at high speed when not wanted. The main difference between flaps and slots is that the latter simply extends the lift vs AoA curve to a high value of the highest coefficient of lift value. When the main wing approaches stalling angle with flaps, it provides steady lift and provides stability

during the rate of climb and which in turn helps the pilot to pitch up more with stability. In aircraft, flaps and slots can be operated individually or combined depending upon the requirement. A pilot can switch the system.

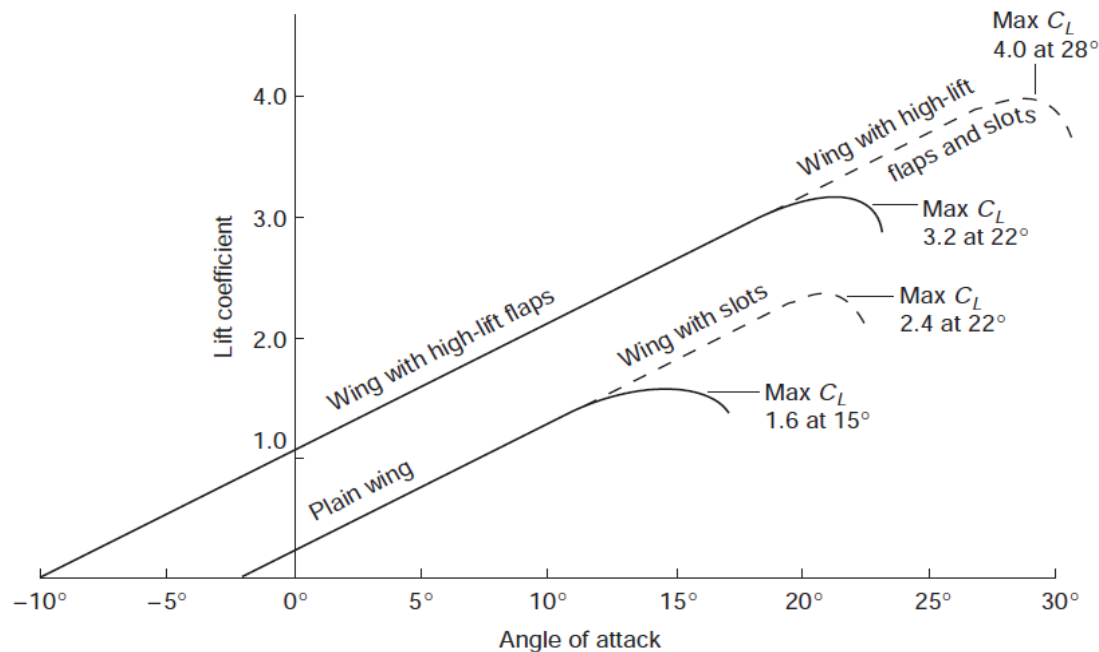


Fig 1-2 lift augmentation effect on the coefficient of lift versus AoA

the simplest of the flaps, such as camber flap, slotted flap and single slotted flap, increase the coefficient of lift and increase in the lift for a normal aerofoil, and consequently, a logical approach of the aircraft intended for landing. They also amplify the drag for landing, which is, for all time, an additional benefit. The additional complex flap, such as the zap-flap and fowler-flap, and the dual and Triple slotted-flap, gives an even better augment in maximum lift coefficient, generally at a bigger AoA, other than basically an efflux of air or jet flow in the shape of sheet of air expelled under force at or close to the trailing boundary of the airfoil, this in fact helps to manage boundary layer, if the mass of the air can deflect the response of the aeroplane will too add straight to the lift force.

A special category of flaps to increase drag is present in aircraft, such as spoilers, air brakes, or both of them together to help the aircraft to slow down after landing. They are used for a variety of purposes on a special type of aeroplanes, to destroy L/D proportion and to further reduce the

angle of glide on sea aeroplanes, to ensure the velocity ahead of turning or manoeuvring, to support equally lateral and longitudinal control, to offer quick pull up after the landing, and in a high-speed aeroplane to reduce the speed from reaching the critical value during dive.

|                                                                                                                              |      |     |                                                                                                                      |
|------------------------------------------------------------------------------------------------------------------------------|------|-----|----------------------------------------------------------------------------------------------------------------------|
|  <p>Slat and slotted flap</p>               | 75%  | 25° | More control of boundary layer. Increased camber and area. Pitching moment can be neutralised.                       |
|  <p>Slat and double-slotted Fowler flap</p> | 120% | 28° | Complicated mechanisms. The best combination for lift; treble slots may be used. Pitching moment can be neutralised. |
|  <p>Blown flap</p>                          | 80%  | 16° | Effect depends very much on details of arrangement.                                                                  |
|  <p>Jet flap</p>                            | 60%  | ?   | Depends even more on angle and velocity of jet.                                                                      |

Fig 1-3 Different types of flaps

### Wing morphing and Variable camber mechanism

Wing Morphing is a potential concept in the field of aviation as it can increase efficiency and remove the limitations of the aircraft. That is the reason why many researchers are trying to design new mechanisms for morphing and also working on the old mechanism with a different approach.

Morphing is not a new concept. It was in the aviation field from the era of the Wright brothers. Morphing was not popular because of its limitation at that time because of material availability, but today we have smart material and advanced technology so that we can incorporate that mechanism into the aircraft and extend the boundaries of aircraft.

With the introduction of variable camber mechanism in the proposed NACA airfoil series, a study is conducted at different subsonic flight regimes, values of coefficient of lift and coefficient of drag are plotted, and increment of lift and drag parameters are calculated with

empirical formulas, and variation of lift and drag values are plotted, and comparative analysis is done with normal NACA series and variable camber NACA series.

## **1.2 Era's in aviation**

The evolution of humankind and the dawn of human intelligence era's in aviation can be divided into three main categories and further can be divided into subcategories as shown below

- i) Early years- 200 BC to 1890 AD
- ii) Pre powered flight- 1860 - 1903
- iii) Early powered flight- 1903 - 1914

In the early years, human knowledge regarding aerial flight mainly originated from closely watching the birds, i.e. flapping the attached wings up and down to create lift. Regarding this, Greek mythology records an event of a father and son duo named Daedalus and Icarus. They believed that they made wings fastened with wax and attached it to their arms to fly and free themselves from prison on an Island, escaping by flying through the air. Irrespective of the warning from Daedalus, Icarus flew closer to the sun, damaging the wings by melting and losing his life. This is the first myth recorded in history related to human flight. So it could be understood from the fact that early human knowledge was based on bird's movement in the air. Various unsung ancient people made wings and leapt from towers, roofs and hills, flapping vigorously and gradually. The idea of strapping the wings fell out of favour.

The idea of attaching a mechanism that could be able to mimic was put forward by Da Vinci. He was possessed by human flight in this regard. He made a few concept sketches of different types of ornithopters towards the end of the 15th century. The figure below shows his conceptual sketches which he has drawn sometime between 1486 and 1490. However, later it was evaluated that human-powered flight by flapping is not enough to lift the human body. G boraeli calculated that the power to weight ratio value of birds is pretty higher than human beings, so it was concluded that humans might not be able to fly with flapping wings, so Da Vinci's contribution doesn't make any contribution towards the advancement of flight. In 1782 Mont golfer brothers

were successful in making their first hot air balloon flight in France and made their first maiden flight in November 1783. With this event, humans were off the ground, but this event didn't bring any significant technical contributions to humans heavier than air flight.

### **1.3 Pre-powered flight**

During the pre powered flight era, the major and most contribution was given by three people whom history speaks about are Sir George Cayley, Octave Chanute, and Otto Lilienthal, The modern aircraft design which we see today is set forth by George Cayley, In 1799 he introduced the idea of the wing attached to the fuselage of the aircraft and brought the concept of stabilizers for manoeuvring of aircraft and even proposed separate mechanism for propulsion.

Sir George Cayley's major contribution to humanity was aeronautics. After introducing the fixed-wing concept of wings in an aircraft, he had his major research on aerodynamic investigation and development. In 1804 for testing airfoils, he built and whirl arm apparatus and designed a fixed-wing glider with adjustable tail plane assembly. George Cayley published the first theoretical and applied aerodynamics article in history. In the year 1849, he tested a full-size aeroplane, and during the test flight, a 10 yr old boy was made to sit in the vehicle and was lifted several meters above the ground while gliding down the hill. Cayley was the first to suggest the use of bi planes and tri planes because he was mainly concerned with the failure of the structural integrity of the aircraft. One of the most important research articles was published in Mechanics magazine in 1852, and the name of the article was "Sir George Cayley's governable parachutes". During his 84 years of life, he founded the basis for practical aviation, and he is called the "father of aerial navigation".

Otto Lilienthal-The Glider man is known for his love and passion for flight. His name and stature have often been compared to Sir George Cayley and Wilbur and Orville Wright. He completed his degree in mechanical engineering and soon developed an interest in flight. In the beginning, he conducted a few experiments on his own designed ornithopters. Later his work and interest turned into gliders with non-movable wings attached to the fuselage. Later he went on to publish a book on aviation that contained data that, at that time, was one of the most valuable

aerodynamic data available as a resource. It is believed that the Wright brothers incorporated some of the data from Lilienthal's book to make their first aircraft, "The Wright flyer 1."

Overall, Lilienthal made more than 2500 successful glider flights, and the aerodynamic data was successfully published as papers and circulated worldwide. However, in 1896, he died during one of his regular glider flights. While flying, he encountered a gust that stalled his glider. He fell to death. There is a feeling that if Lilienthal lived, he must have beaten the Wright brothers. After these aviators, we see those aeronautics came to the united states of America. At the time, the nation was busy with new governance and expanding its frontiers; a new aeronautical enthusiast was spotted in Chicago. Octave Chanute, a civil engineer by profession, somehow got interested in mechanical flight and over the span of 35 years, he collected and preserved every piece of aeronautical information, and this resulted in the publication of a book named Progress in flying machines in the field of aeronautics. This book earned him the title of the first serious aviation historian. Chanute's positive suggestions were incorporated by the Wright brothers, which led to the first powered successful flight in 1903. Chanute was an airman, which led him to experiment with hang gliders. His major work contribution was the successful design of a bi plane hang glider, which introduced the Pratt and truss type method of structural rigging.

In the same time frame, two other aviation pioneers were at work tirelessly building and designing a series of powered aircraft, which resulted in successful powered flights in 1903. Wright brothers and Samuel P Langley were the other two aviation pioneers who later developed an interest in powered flight experiments.

### **Powered flight**

Samuel P Langley followed the examples of George Cayley built and started experimenting with different shapes of Aerofoil on a whirling arm apparatus, his earlier works were on steam-powered model aeroplanes, and he called them aerodromes. Aerodromes were basically tandem wing aircraft, and with the success of steam-powered aerodromes, Langley decided to work forward towards a piloted aircraft which he got funding from the US government itself a sum of 50,000 US dollars.

He made two attempts with his aerodromes with the pilot; both of his attempts met with failure, and it is believed that it was a structural weakness that led to collapse, and he completely ignored

the important aspects of flight control, his pilot Charles manly has no flight experience. Nine days after Langley's failure Wright brothers recorded their first successful flight. The Wright brothers are regarded as the premium Aeronautical Engineers, although by profession, they were bicycle manufacturers. Only George Cayley can be considered to compare with the Wright brothers. The achievements of the Wright brothers are considered to be more valuable than anyone in the field of aeronautics.

Wilbur and Orville Wright are considered to be the first practical aeroplane inventors. They had a college diploma and had business opportunities which prompted them to manufacture their own bicycle designs using homemade tools; their business was profitable and helped to provide the financial backing for their aeronautical ventures. It was Wilbur Wright who took the study of birds flight as he believed it would lead towards mechanical flight; they brought and possibly read all the data available at the time and worked on designing wing-warping, which they tested and found to be successful which they believed led them to successful controlled glider flight demonstration. During the course of time, they developed a good relationship with Octave Chanute, which benefited both parties. Overall, before the successful flight, the Wright brothers made over 1000 perfect flights, and in the process, they became very successful, highly skilled professional pilots. And in 1903, on December 17 Wright brothers had their first successful flight.

### **1.3.1 Future possibilities and innovation in Aviation**

According to ICAO (International civil aviation organization), Aviation will be the main engine of development; Innovation & technology will make engines and aircraft lighter in weight, the operation will become much better more efficient and will change the entire scenario, Newer technologies like wide-body aeroplanes which could accommodate more passengers and blended wing body fuselage concept (project nick named maverick) by AIRBUS already into conceptual design phase both the concept deliver 25% saving in fuel consumption and carbon dioxide emissions when compared to previous generation aircraft.

Similarly, wing morphology, shape-shifting wings and variable camber mechanisms are new concepts that could change the way aeroplanes are used to fly; by the introduction of AI, the availability of new structural materials will further enhance the scope of aviation in nearby

coming future. Nowadays, human factors are also taken into consideration in aviation while creating cutting edge technology. With more accessibility, aviation has become the number one choice for the mass population as; if now 400 flights depart every hour too, only counting commercial aviation; the major challenge is that quickly airspace is going to be congested in order to accommodate the growth drastic changes and improvements need to be found that is we need innovations and new technology development, As we know that aviation is the driving force of global technology development and innovations, Engines and aircraft become lighter, quieter and more efficient, Emerging technologies are reshaping with robotics, artificial intelligence, the internet of things, unmanned aircraft systems and the push for hybrid and electric aeroplanes, alternative fuels have already changed the current scenario of aviation

#### **1.4 Origin of aerodynamic forces on a wing section**

If we want to know how an aeroplane flies, then we must have a clear idea of and the difference between the engines and the wings the different work they perform. An aeroplane's engine is made to move the aircraft at a high forward speed. This makes the air flow move on wings creating the required pressure difference on the mainplane; hence it can be explained that according to Bernoulli's principle, the upward force, i.e. Lift, is obtained on the wings mainly due to change in pressure parameters. Although Bernoulli's principle is correct, it fails to explain why the curved surface on the top has less pressure, and in practice, we have seen aeroplanes capable of inverted flight.

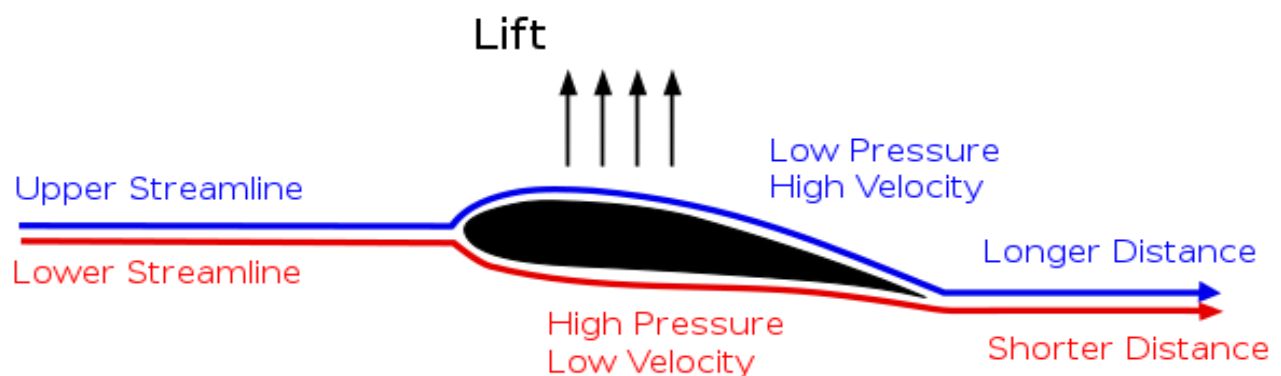


Fig 1-4 Flow over an airfoil

Newton's third law suggests that wings' downward push results in an equal and opposite reaction which creates lift; this Newtonian concept can apply to any shape, curved or flat, symmetrical or

not, and besides all, this law holds good for inverted flights also. But then again, it fails to explain why there is a low-pressure region atop the wing irrespective of whether the airfoil is cambered or not. Liu et al.; constructed the complete theory of Lift generation to understand and visualize the source of lift production. In viscid circulation theory, Lanchester was the first to introduce a theory of vortex creation on wing tip for creation of lift, and from there, the famous Kutta joukowski theorem is introduced where an important physical quantity  $\Gamma$  is introduced.

$$L = \rho v \Gamma$$

where ( $\Gamma$ ) in the formula stands for circulation around the wings. There are other theories explaining lift, but none holds good as the lift line theory. Therefore moving with a viscous theory of circulation is the next forward way to deeply access the generation of lift around an airfoil.

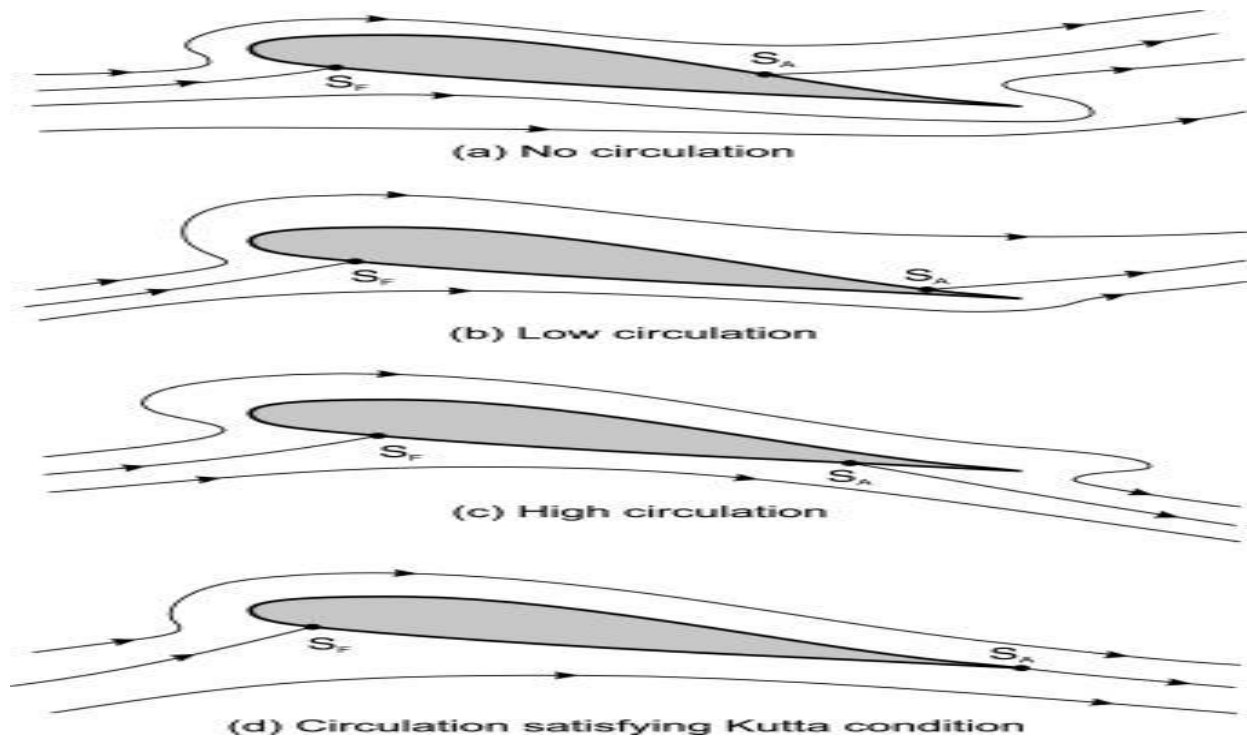


Fig 1-5 Kutta condition for different flow circulation around an airfoil

## 1.5 History of airfoil development

Development of airfoils started around the 1800s; at the time, it was known to researchers that flat plates kept at an angle of incidence produced lift; later, they believed that shapes with curvature resemble more bird profile wing shapes, and HF Philips was the first person to patent aerofoil design and got experimental results in early wind tunnel experiments. In the year 1893, Octave Chanute predicted extensive studies must be conducted on various cambered and symmetrical airfoil structures. Otto Lilienthal had similar views about airfoil structure. Airfoils used by the Wright brothers were very thin and highly cambered; the reason for this assumption was due to testing of airfoils at low-speed wind tunnels, which was the only availability; this led to a belief that aeroplanes may require more the one wing, that's the reason almost all the aeroplanes during that era were bi planes. A wide range of airfoils was manufactured on a trial and error base, as shown in the figure below.

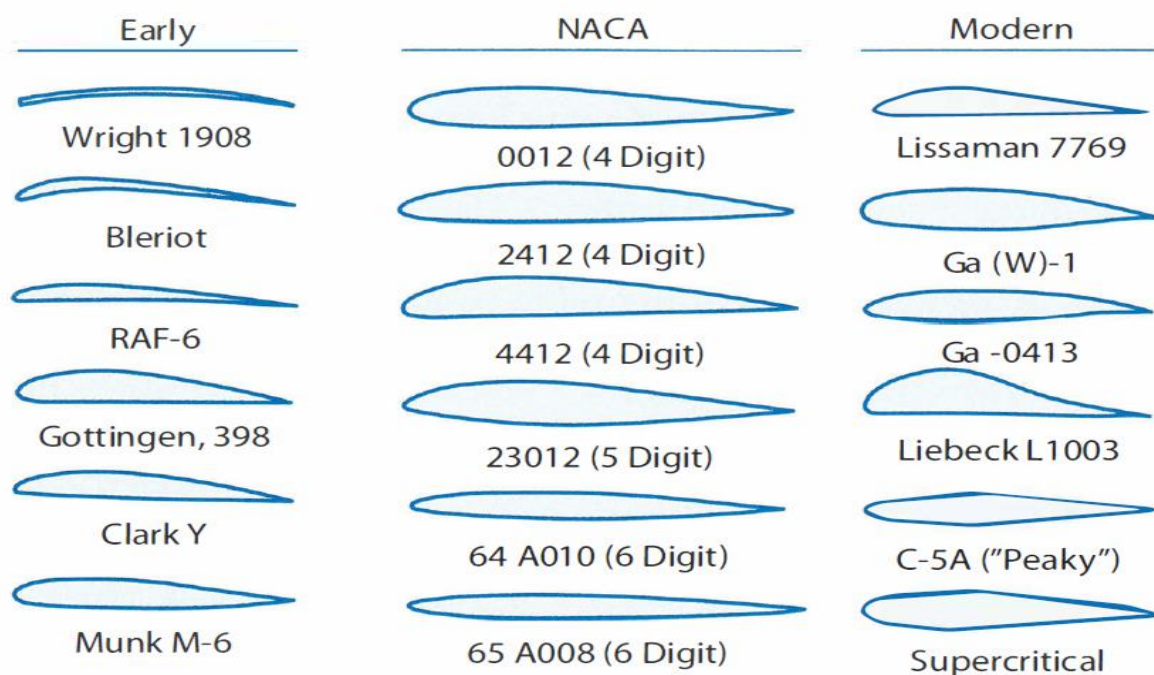


Fig 1-6 Airfoil series

By varying the design parameters such as thickness and camber, some successful enormous results were observed named clarkY and Gottingen and tested by the NACA series in the 1920s. In 1939 for the first time, laminar flow airfoil sections had done by Eastman Jacobs; this research

outcome a very low drag and lift to drag ratio of 300. This concept of Jacob was not accepted fully because of the consideration of the Reynolds number. The performance of the airfoil changes based on the flow and Reynolds number, so the design of the airfoil also changes according to the requirement; this is the main reason that the modern era has no best airfoil yet because of design parameters change based on the requirement. Sometimes based on the requirement, the design constraint may vary and leads to unconventional shapes; for example, in the case of an ultra-light sailplane, the requirement is small pitching moments, and large maximum lift coefficients at high speeds situations like this result in a drastic variation in design but then yes, we have a possible solution for this that is using a wing which could vary camber.

### **1.5.1 Airfoil geometry**

The geometry of the airfoil is characterized by the parameters such as maximum thickness, camber, and their positions along with nose radius; these parameters are basic requirements for designing an airfoil. The NACA 4 and NACA 5 Digits were created by a single mean line profile with varying thickness distribution along the chord line.

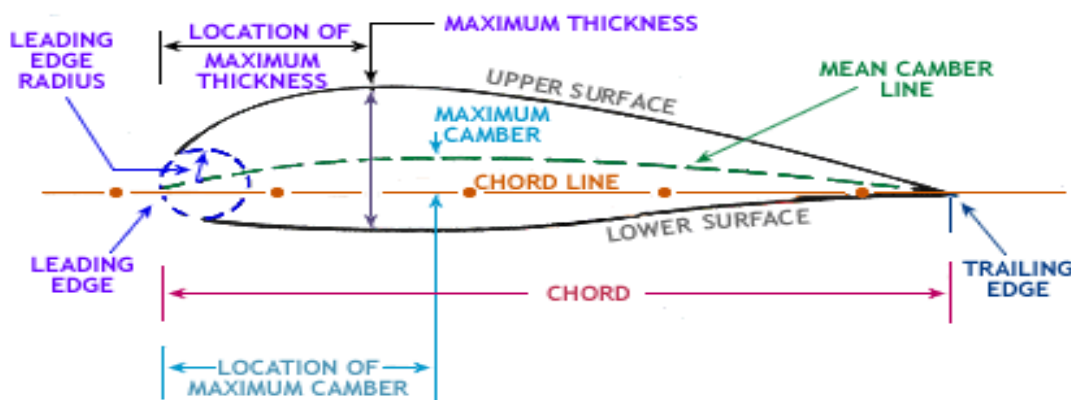


Fig 1-7 Airfoil geometry

The camber line of 4 digit NACA series is two parabolas. The four digits of this series have a specific significance of parameters – the first digits tell us about the max camber % in the chord, the second digit gives us an idea about the camber max position that is one-tenth of the length of the chord the last two digits represents and defines the max thickness in % of chord. NACA 5

digit is like NACA 4 digit; the only variation is camber line has more curvature at the nose. NACA 6 digit series and above are intended to achieve laminar flow at high speeds. These series selections are specialized for the application. Efficiency, performance, low drag, and maximum lift capacity are required for designing the wing and other parts of the section in 3-Dimensional.

### **1.5.2 Airfoil pressure distributions**

Pressure Distribution around the cross-section plays a significant role in generating lift. When the flow is curved, we must have a pressure gradient in the direction of curvature coefficient of pressure is negative. An airfoil with a reflexed curve profile has less camber over the aft and results in poor lift generation and low nose-down pitching moment and coefficient of the moment is positive in supercritical airfoils aft part contains more lift.

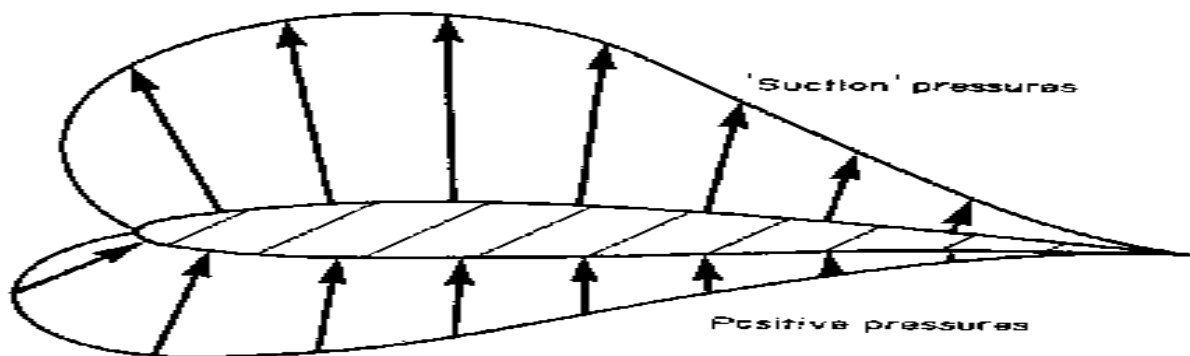


Fig 1-8 Pressure distribution on an airfoil

The Design of airfoil matters based on our own requirements such that sometimes we design to produce low drag, sometimes low drag with high lift, and in a few cases, we do not bother about lift; we are concerned about aerodynamic parameters, and among these, the main agenda is to achieve good performance with constraints such as pitching moment, thickness or off-designed parameters. To design a new sections/airfoils, it is important to cite previous data for designing the airfoil; we have two methods, Direct and Indirect methods.

### **1.5.3 Airfoil performance**

The direct method deals with the geometry of the section along with pressure and performance parameters. Using this method, we can change the shape to improve performance; on the other hand, this method has drawbacks too

- i) identification of performance measure
- ii) method to approach to change the shape to get better performance than before.

In this Direct method, first, we will assume airfoil shape mostly from the NACA series; this method helps to define the characteristics, which is the major concern in all design models. The design of an airfoil does not require a specific scalar objective function but is essential to identify the potential problem to make sure without committing any errors. Sometimes the aim of airfoil design is to fix the worst things, such that drag reduction at high-speed regimes by maintaining maximum lift coefficient to achieve this slowly gets increased, effect on maximum lift should be noticed drag coefficient gets reduced by using the constraint of maximum lift coefficient, i.e.  $(C_l)_{\max}$  and also lift to drag ratio can be maximized it plays an important role and it is important to choose wisely it must be considered with respect to the aeroplane, not by the consideration of only wing section.

### **1.5.4 Methods of airfoil design**

The simple method of an aerofoil design is to start with an assumed profile shape and then the calculation of pressures and performance. The inverse Design method is another type of Design method; the main aim of this method is it is much concerned with the distribution of pressure. It's necessary to have a proper distribution of the centre of pressure to get a minimum pressure coefficient. Based on the results, we use the thin airfoil theory for the shape of the camber line with the detailed distribution of pressure differences in potential flow. In this method also, changing shape to get better performance is the main theme; it can be done In many ways

- i) by observing the effect of geometry changes on  $C_p$  and its performance
- ii) by using shape functions to improve the design performance.

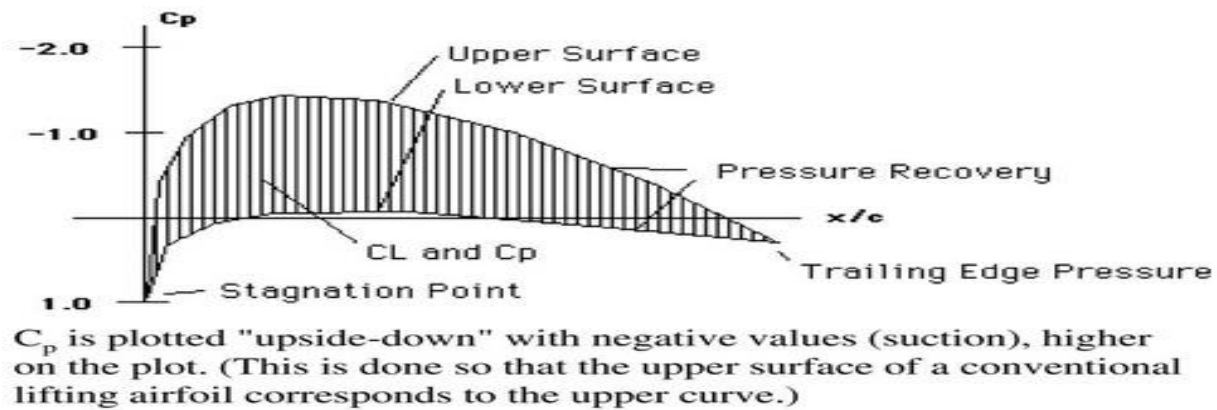


Fig 1-9 Pressure distribution around an aerofoil

### **1.5.5 Airfoil design problems**

A typical airfoil design always encounters some common; here, we will be discussing some common problems. The major problem of designing thick airfoil is that the value of minimum pressure is reduced due to the change in profile thickness and hence induces a gradient of pressure, so we need to start recovery soon. Another design parameter is Laminar flow Design, where laminar flow is useful for the reduction of a drag caused due to the friction by the skin of the aircraft called Skin friction drag, to maximize achieved Lift and to reduce the transfer of heat at low Reynolds number with a smooth surface with a favourable gradient of pressure. Another design is Transonic airfoil Design; in this design, the difficulty arises on shock drag minimization, which limits the minimum pressure coefficient; as we know, lift and thickness reduce the minimum pressure coefficient in transonic design we need to design airfoil in such a way that it can have high lift and thickness without causing strong shockwaves. Supersonic flow is achieved without drag increment with The design of supercritical airfoils, and local Mach numbers for this design should not exceed 1.2-1.3.

In supersonic airfoil design should follow some aspects lift should be more in the aft position along with sufficient lift in the upper surface of the forward position pressure peak near the nose of the airfoil section diminished as the Mach number increases lift in the region. The lower surface of the supersonic airfoil cross-section near the L.E leads to a positive pitching moment and lift. Shocks in the front section of the surface produce less wave drag than the aft section, so

peaky airfoils are used on many transport aircraft. For the design of supercritical airfoil, several parameters need to be considered.

- i) too much aft load leads to the greater value of a negative pitching moment.
- ii) manufacturing of thick trailing edge is difficult
- iii) In laminar flow boundary layer can handle only gradual pressure recovery.

## **1.6 Different types of wing designs**

Throughout decades different types of wing configurations have been designed and tested; in history, some of the tested designs were successful; here, we will discuss different configurations and their pros and cons. The main plane of an aircraft is basically the shape of an airfoil which creates an upward force that is lifted when moving through air. Engineers and design analysts have tested different wings for various flow parameters which govern the flow around a wing section; the wings or main plane of an aircraft can be fixed at a different position on the fuselage of an aeroplane. In this section, we will discuss types of wings and different wing positions and structures of aircraft wings.

### **1.6.1 Wing configurations**

Aircraft are classified according to their wing configurations; fixed-wing aircraft can be fitted with a different number of wings, such as triplane, bi plane, and mono plane and can be mounted at different positions and at different angles to the fuselage, from the 1930s onwards we only see monoplane configuration. The angle at which they are attached is called dihedral angle and anhedral angle, and it affects the aircraft's lateral stability.

### **1.6.2 Wing structure**

Wings of the aircraft help to make the aeroplane airborne; any design of wings for an aeroplane depends on several critical performance parameters such as wing loading, take-off speed, landing speed, rate of climb etc. Nowadays, the wings are monoplane, and it's a complete cantilever which means this type of wings doesn't require any kind of external support. Materials used in airframe construction must possess some airframe material properties such as high strength to weight ratio, light weight, corrosion-resistant, should be non-flammable, and must possess high

quality. Few materials we use in airframe construction are wood (spruce), steel and its alloys generally for they are strong, one of the common materials used is aluminium, and its alloys, titanium alloys are used for heat barriers earlier generation typical supersonic aircrafts used titanium for building the airframe in order to sustain skin friction generated during supersonic flight.

Magnesium alloys are also used because it is three times lighter than aluminium. For the last few decades the, researchers and the manufacturers have extensively done research on plastic and its components, and even we are aware that composite materials have replaced traditional alloys which were used in making the fuselage; Boeing 787 dream liner is a very good example as the 50 per cent of the aircraft wing is made of composite reducing the weight of the aircraft nearly half, reducing the empty weight of an aircraft which is almost desired and preferred by any aircraft manufacturer. Construction of fuselage structure can be done by various methods; there are three types of fuselage construction truss type, monocoque and semi-monocoque type construction out of which semi-monocoque type construction is preferred over other constructions. Most early-type aircraft used truss type construction; mostly warren truss type construction was used to manufacture fuselage. Semi-monocoque type construction has different components like skin, formers, stringers, and bulkhead. The loads are distributed among vertical members and longitudinal members. Load acting on the aircraft first acts on the skin of the aircraft, then it is distributed to formers through longitudinal component, and ultimate concentration of the load comes to the bulkhead; every aircraft nowadays employ semi-monocoque type construction.

### **1.6.2 Wing position over the decades and its progress**

The wing may be considered the most important component of an aircraft since a fixed-wing aircraft is not able to fly without it, and its geometry influence all other aircraft components. The primary function of the wing is to generate sufficient lift force. However, the wing also produces drag as well as pitching moment; Always; a wing designer is looking to maximize lift and minimize drag due to lift (Induced drag). Basically, the principles and methodologies of system engineering are followed in the wing design process. A single wing usually has a longer wingspan compared with two wings; Old manufacturing technology was not able to structurally support a long wing, with advancements in technology and with the availability of strong

aerospace materials. A few other design considerations may still force the modern wing designer to lean towards more than one wing.

### **1.6.3 Rectangular wing design**

The rectangular wing is used in small aircraft and is one of the easiest to manufacture. These wings have the same chord length from root to span of the wing, These wings don't have a proper elliptical lift distribution, but if a twist is provided along the span, it will almost match the lift distribution of an elliptical wing. A rectangular wing has a natural tendency to stall from the root; a rectangular wing and tapered wing almost possess the same features, with nothing much difference; both could be used for high-speed aeroplanes. In the early 20s and 30s, from float planes to fast fighter aircraft, they all used rectangular wings. The advantage of a rectangular wing is that its wing ribs are interchangeable.

### **1.6.4 Elliptical wing design**

Frederik Lanchester, a theoretical aerodynamicist was the first to research in detail elliptical wings; elliptical wings were used during the 1930s and 1940s and went on for mass production during the above mentioned period. The elliptical wing design was created to occupy ammunition guns and landing gear later; it proved to be one of the best designs which provided better aerodynamic performance in the air than any other wing. This wing design was capable of producing uniform lift distribution, with improved stall characteristics, elliptical wing designs which are slightly twisted and made thin towards the tips, will give washout, and the inner wing will stall first, which is an added advantage as the ailerons will be active even a case of a stall is encountered by the pilot. Initially, the concept of the elliptical wing was regarded as the theoretical one and was not considered for practical application; there are few aircraft that use elliptical wing configuration; one of the famous aircraft to use an elliptical wing is the super marine spitfire used in the battle of Britain. The only disadvantage of this wing is that it is difficult to manufacture due to its elliptical profile. And there is no doubt that these wings look more aesthetically beautiful than any shape profile.

### **1.6.5 Tapered wing design**

These types of wings actually evolved from changing the profile of a rectangular wing. A taper can be provided on one side of the wing or on both the leading and trailing edge, and a tapered wing is not efficient as the wing of a Super marine spitfire, which has an elliptical wing. If we plot a lift curve along the span of the wing, it will closely represent elliptical lift distribution along the span. The Mustang P52 was a legendary aircraft that incorporated a tapered wing and was aerodynamically superior among the aircraft flying during that era. Mustang aircraft were manufactured by North American aviation.

### **1.6.6 Delta wing**

Delta wings are considered one of the best all-rounders in all flight conditions. That is, it is a performer even the aircraft is moving in subsonic, transonic and supersonic flight regimes, aircraft with delta wing are manoeuvrable and provides manoeuvrable high-speed capability; these types of the wing are much stronger and are capable of storing a large volume of fuel, these type of wings are simple to manufacture. Mirage 2000, manufactured by Dassault, is a good example of delta wing aircraft; similarly, the Euro fighter typhoon used cropped delta wing. Another variant is found in the double delta wing.

### **1.6.7 The trapezoidal wing design**

This type of wing used in combat aeroplanes has one of the best outstanding performances, the end of the main plane sweep forwards, and the front end of the main plane sweeps backwards. This type of wing is capable of flying efficiently at supersonic speeds and has stealth characteristics; the only disadvantage of this type of aircraft is that it has high wing loading. F 22-A Raptor, a US combat aircraft, uses this type of wings

### **1.6.8 Ogive wing design**

The legendary Aerospatiale-BAC-Concorde used Ogive wings; these types of wings were used in high-speed commercial aeroplanes very difficult to manufacture them is difficult, and moreover, their subsonic performance is not satisfactory in comparison since these types of

wings are good only at very high speeds. Special attention was provided during the manufacture and design of these wings when compared with other types of wings; wings of a subsonic aircraft will have more than 50 control movable devices, which are used to move flaps, slats, slots; these are used for additional lift during takeoff and landing of an aircraft and to control and trim, the Concorde aircraft slender wing doesn't have these type of control devices. It only has six elevons, capably replacing traditional controls. The ogival wing produces lift at a wide range of angles of attack, so the stalling of these types of wings was delayed to a higher angle of attack.

#### **1.6.9 Swept back wings**

Swept-back wings are the most common wing configuration used in transonic and subsonic commercial jet aeroplanes. During the transonic flight, these type of wings delays critical Mach number by splitting the airflow into two components, namely spanwise and chordwise flow; by doing so, it delays the formation of shock wave formation on wings. Sweep back wings allows a higher value of critical Mach number compared to the rectangular and tapered wing. The advantage of the swept wing is that they delay the formation of wave drag. Swept-back wings give us directional as well as roll stability. B787 dream liner manufactured by Boeing is one of the famous aircraft which uses a swept-back wing configuration.

#### **1.7 Wing Morphing mechanism**

Future aircraft needs technologies which could increase aircraft versatility and utility, An aircraft capable of morphing its wing shape, creating radical change in an in-flight profile will be able to survive a real threat scenario. Morphing wing concept through which we could achieve radical shape changes. Over the decades, the morphing concept has brought a radical change in order to enhance performance characteristics in different environments. Researchers in the aviation field are desperately looking for other ways to increase aircraft efficiency by maximizing the Lift to drag ratio and increasing aerodynamic performance. Many times, history are the best answer for future problems, so scientists started looking for the mechanisms in the past and then started solving their concerns with advanced tech. The primary purpose of these researches is to make fixed-wing highly efficient. The primary motivation for morphing is birds, as they use the same wing efficiently in various conditions. Wing morphing is not a new concept. It was in the field from the time of the Wright brothers, and they named it to wing warping as the current aircraft

are unable to fulfil the conditions of upcoming problems, so there is a need for change. The wings are built as a compromise geometry that allows a plane to fly in various environments, but performance is always sub-optimal in each. Morphing is a term used in the aerospace industry to describe a group of technologies that improve a vehicle's performance by modifying specific characteristics to better adapt the vehicle's condition to the environment and mission at hand [1]; on the other hand,,, morphing is short for metamorphosing. Morphing is categorized in three sections as follows.

Mimicking birds to enhance the aerodynamic efficiencies of modern-day aircraft is a method that has seen major applications in the aerospace industry. Birds, due to their feathers, muscles and distinct morphologies, tend to utilize methods that are otherwise impractical to incorporate in aircraft due to limitations in terms of structure and materials. Yet, major influences are seen in many aircraft that are inspired by the aerodynamic features of avian beings. Morphing is a major method that helps aircraft make changes to their components, such as their wings, to adapt or adjust to their varying flight conditions for efficient control and performance. In this paper, a mechanism is designed that aims at using morphing techniques to mimic a bird's tail. The main purpose of this design is to achieve yaw control and directional stability in the absence of a rudder and with the aid of the horizontal stabilizer solely

### **1.7.1 Types of morphing**

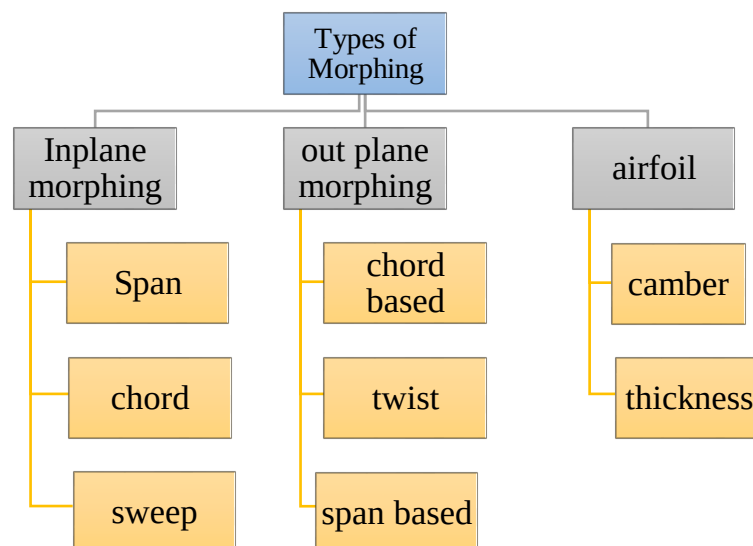


Fig 1-10 Types of wing morphing

### **1.7.2 In-plane Morphing**

In-plane morphing cooperates with 2D movement, and it consists of –

- Variable sweep
- Variable chord
- Variable span
- Co-planer joined wing

#### **1.7.2.1 In-plane span wise morphing**

Variable span morphing is the type in which we can manipulate the span owing to different mechanisms. Variable span is a famous in-plane morphing type that has been used and tested since the 1930s. The use of variable span can eliminate the problems faced by both these wings and can in cooperating their advantages together, i.e. single aircraft or UAV with the ability to shrink or enlarge its span of the wing according to the need.

Table 1-1 Difference between Long span wing and short span wing

| Large span                                                                            | Short span                                                                       |
|---------------------------------------------------------------------------------------|----------------------------------------------------------------------------------|
| i) Good range<br>ii) Fuel efficient<br>iii) Less manoeuvrable<br>iv) Low cruise speed | i) Poor range<br>ii) Not efficient<br>iii) Highly manoeuvrable<br>iv) High speed |

Table 1-2 Critical performance parameter evaluation of variable span wing

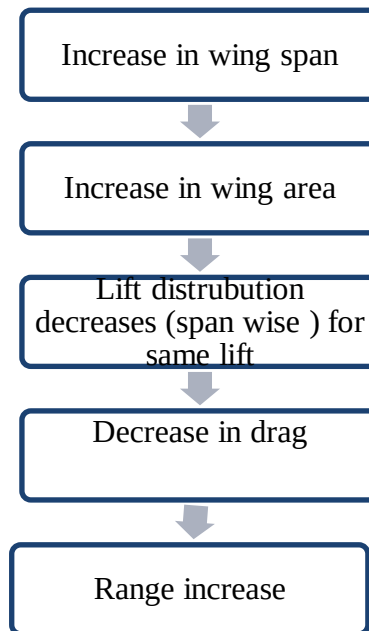


Table 1-3 Advantages and disadvantages of variable span

| Advantages                                                                            | Disadvantages                                |
|---------------------------------------------------------------------------------------|----------------------------------------------|
| i)Increase in endurance and range after extension.                                    | i)Increase in weight                         |
| ii)Take-off and landing distance increases                                            | ii)Risk of mechanical failure or malfunction |
| iii)Flaps can be avoided as we can use an extension of the wing to increase the lift. |                                              |

### The mechanism used for Variable span morphing

#### ➤ Mechanism 1. Telescopic wing

This mechanism consists of two separate wings, one wing inserted in the other, and a tool to manipulate the span of a wing by moving these wings one over the other.

Some examples of Variable span aircraft (fixed wing)

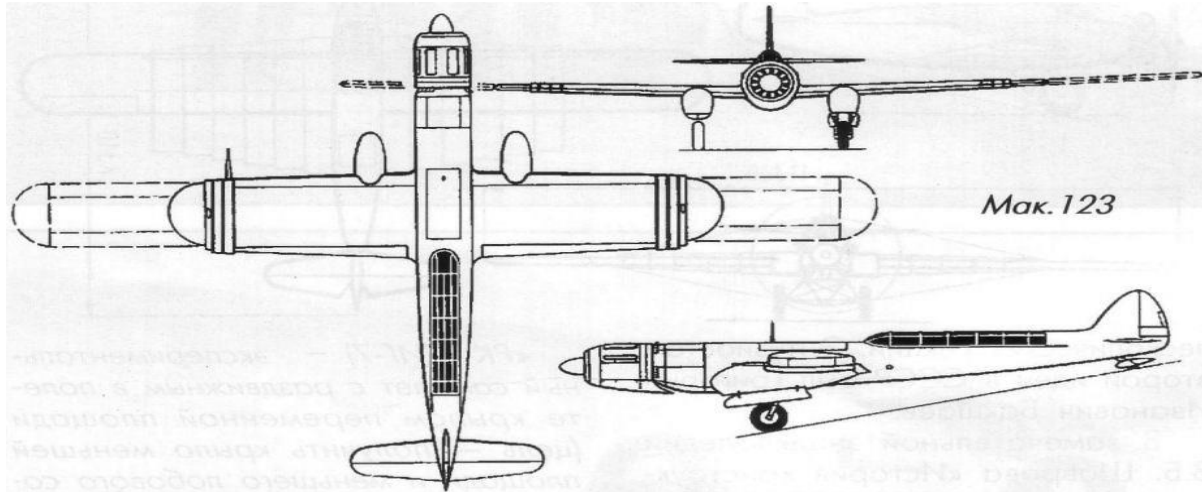


Figure 1-11 Mak 123 aircraft with variable span mechanism

### Research and experiments in variable span

The company named Gevers Aircraft Inc. from the USA in the year 1997 researched the telescopic wing by cooperating with six-sealer amphibious aircraft. The wing was divided into sections centre was a high-velocity wing that allowed to lower aircraft speed while landing and allowed to increase the aircraft velocity while in cruise mode. It was able to land on hard surfaces, snow, and water. The telescopic wing was designed and then fabricated by Blondeau.[2] The wing was three-segmented, and the material used was hollow fibreglass.

### The following points were observed during experimental flights

- The aspect ratio can vary around 114%
- L/D ratio after extension achieved 9 to 10
- The wing section was also the reason for the parasitic drag

In the year 2007, this project was modified such as [3]

- Use of two spars that were identical
- AR (Aspect ratio) can vary around 230%
- Parasitic drag was also reduced by decreasing the height.
- L/D max =16

Henry and Pines analyzed (2007) that an increase in span can be effective in increasing the lift or manipulating the Aspect ratio and being effective in roll control. A perfect moment can be achieved in the span extension than in aileron deflection. Damping in the system increases in the variable span more than in the aileron mechanism.[4] Other examples are as follows. In 2003 Arrison and the team designed a fixed-wing UAV for a variable span mechanism, and the purpose of the project was to increase the L/D ratio as required. The skin used for the UAV was sliding, and the actuation used here was rack and pinion.[5]. In the year 2003, researchers Sullivan and Watkins made an aircraft of the size of a UAV with variable span and oblique wing mechanism. The primary purpose of this research was to increase the performance of aircraft by increasing the lift to drag ratio. They used an electromechanical actuator for varying the length angle of attack of the wing. The sliding skin of aircraft also helped the research. In 2005 Heryawan, along with the others, made a Fixed wing MAV for increasing range and performance and the lift to drag ratio. DC motors were used to actuate the mechanism of variable span. The skin of the aircraft was rigid.[6]. Bye et al. and Ivanco et al. 2007 designed and tested an aircraft of the size of a UAV

### **Mechanism 2. Folding wing**

The entire wing folded, similar to the origami technique, and extended when needed to increase the span. Material selection is a very crucial part of this mechanism. Agile hunter by Lockheed martin. The prototype of agile was called as Z wing. This military UAV was able to fold its portion of the wing(inner) to reduce the wingspan, area, and drag of the UAV as well as during transonic flight at low altitudes.[7]. Challenges faced in this UAV. Hinges connect the different portions of the wing because hinges must be strong and stiff to bear the forces acting on it and also provide streamlined flow, and material selection was quite difficult to achieve.

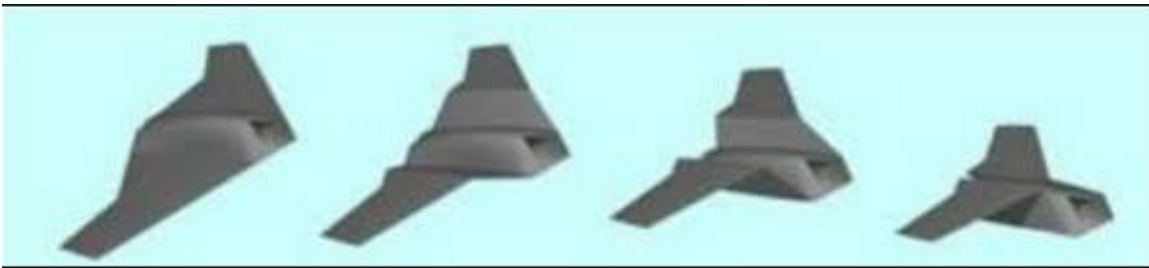


Figure 1-12 Z wing morphing

### 1.7.2.2 In plane chord wise morphing

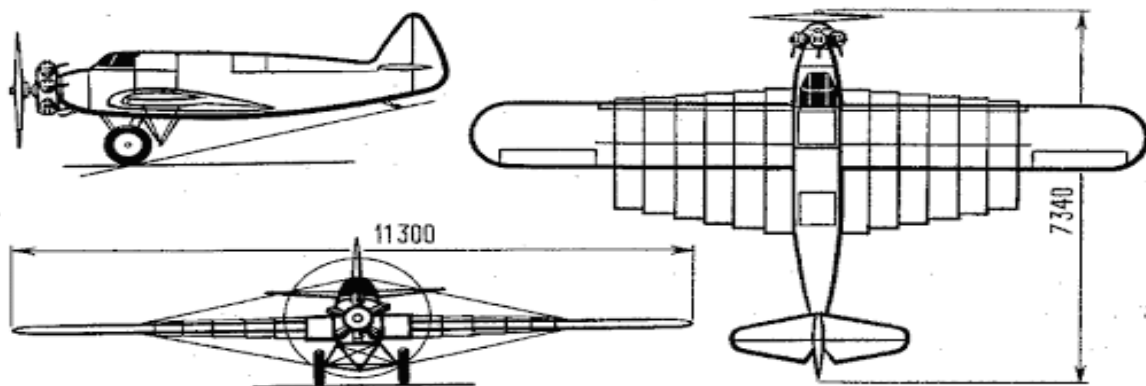


Figure 1-13 Bakshaev LIG-7 aircraft with variable chord mechanism

Chord morphing allows us to manipulate the shortest distance between the leading edge and trailing edge, i.e., a chord of the wing, with the help of some mechanism. We can change the aspect ratio, and area of the wing, which helps us to increase aerodynamic efficiency.

#### Aircraft and experimental research on variable chord

Bakshaev LIG-7 (1937) In 1937 USSR designed a plane that was capable of changing its chord with the help of a telescopic mechanism. It was the first variable chord aircraft. In the fuselage, sections of wings were hidden, and steel wire was used as a mechanism to extend the wing. There were six hidden sections. The area could be increased by 44%. [8] In the year 2005, Reed et al. tested variable chords by rib mechanism to manipulate the chord. They used lead screws actuated by DC motors (miniature) and a rib with a slotted box through which the structure could slide. The help of a flexible honeycomb maintained the airfoil shape. [9]

The challenges faced in this project were

- Forces acting and smooth working of mechanism despite these forces
- Increase in weight due to design
- Material selection

Perkins et al. in the year 2004, used Shape memory polymer for the chord morphing; they tried to achieve an 80% of chord increase.[10] The material selected was Dynamic modulus foam (DMF) was considered for the research as it is a lightweight form of SMP. Temperature actuation was selected for changing the chord. Actuation was done with nichrome wire in the skin.

### **1.7.2.3 In-plane sweep back wing mechanism**

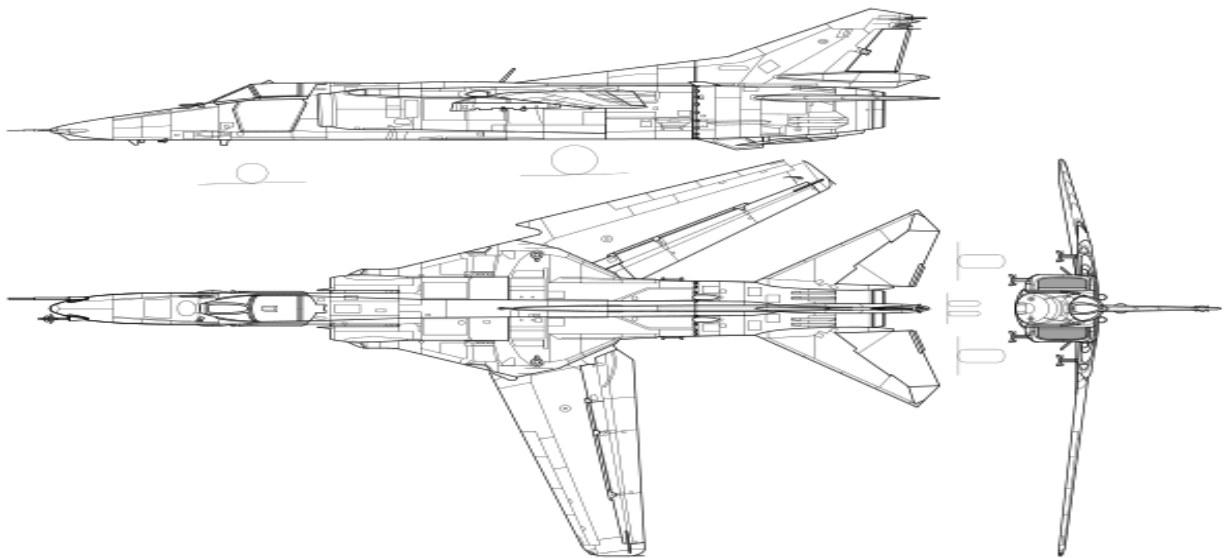


Figure 1-14 MIG 23 with the variable-sweep mechanism

Variable sweep morphing is popular and has given more favourable outcomes compared to other wing morphing since 1931. This morphing allows us to change the sweep angle of the wing shape of the entire aircraft, which helps aircraft in various difficulties. Problems that variable-sweep can solve, Sweep back wing cause problems to aircraft while landing, Sweep back wing face major issues in high-velocity winds, especially in the sea, as they have to deal with turbulence and lack of stability.

Table 1-4 Advantages and disadvantages of variable-sweep morphing

| Advantages                                    | Disadvantages                         |
|-----------------------------------------------|---------------------------------------|
| i) Reduction of take-off and landing distance | i) Increase in weight in the wing box |
| ii) Increase in load-carrying ability         | ii) Increase in maintenance           |
| iii) Simple mechanism                         | iii) Risk of mechanism malfunction    |

### Co-planer Joined Wing

It is the type of morphing in which the design of the wing is not constant or fixed and it is unconventional.

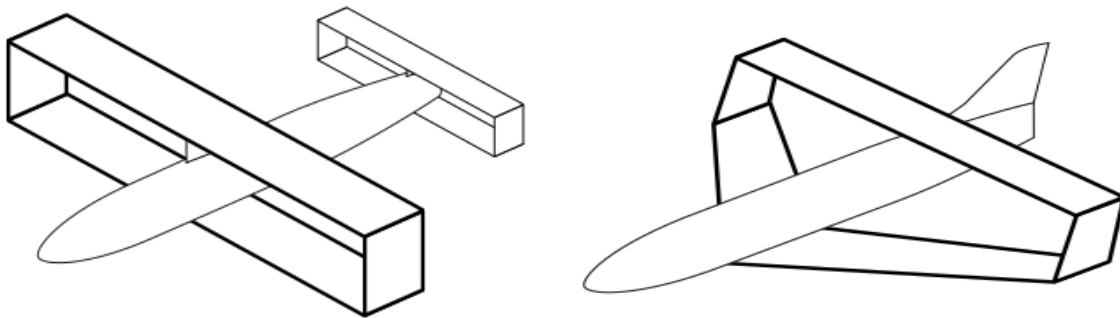


Figure 1-15 Co-planer wing design

Some of the observations on the Oblique wing [11]

- i) Rotational Angle – 0° to 60°
- ii) Low speed – No need to rotate as the base design is sufficient to fulfil the basic needs of the aircraft
- iii) High speed – Drag reduction by rotating the wing at a certain angle
- iv) Aerodynamic efficiency at high speed

Table 1-5 Advantages and disadvantages of oblique wing

| Advantages                                                                                                                                                                                                                                                           | Disadvantages                                                                                                                                                               |
|----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| <ul style="list-style-type: none"> <li>i) Ability to increase the efficiency at high speed</li> <li>ii) High Endurance</li> <li>iii) Noise reduces</li> <li>iv) Drag reduction ability</li> <li>v) Mach increases effectively</li> <li>vi) Fuel efficient</li> </ul> | <ul style="list-style-type: none"> <li>i) Can Rotate in one direction only</li> <li>ii) Less efficient for the low speed</li> <li>iii) Risk of mechanism failure</li> </ul> |

#### Inflatable Wings

Wing in this type of aircraft are flexible and attain a definite shape after being blown. Weight reduces enormously, and this is appropriate for UAVs

Table 1-6 Advantages of disadvantages of inflatable wings

| Advantages                                                                                                                         | Disadvantages                                                                                                                                                                                                  |
|------------------------------------------------------------------------------------------------------------------------------------|----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| <ul style="list-style-type: none"> <li>i) Reduction in weight</li> <li>ii) Portable</li> <li>iii) Reduction in Workload</li> </ul> | <ul style="list-style-type: none"> <li>i) Requirement of smart material</li> <li>ii) Risk of destruction by sharp material</li> <li>iii) Cannot use on large planes and in harsh weather conditions</li> </ul> |

### **1.7.3 Out plane Morphing**

Out Plane morphing is three-dimensional morphing, and it consists of

- i) Twist Morphing
- ii) Spanwise Bending
- iii) Chord morphing

#### **Twist Morphing**

It is the world's first form of morphing, bought and cooperated by the wright brothers in their plane (termed as wing warping). As time passed, this morphing lost focus and production because unavailability of suitable material and instability caused.

Table 1-7 Advantages and Disadvantages of twist morphing

| Advantages                                                                              | Disadvantages              |
|-----------------------------------------------------------------------------------------|----------------------------|
| i) Decreased weight<br><br>ii) Improves L/d ratio<br><br>iii) Increases manoeuvrability | i) Requires smart material |

#### **Span wise bending**

Span-wise bending allows us to reduce the span of the wing and manipulate the geometry in three dimensions. This morphing is very useful in both low as well high speeds as it decreases drag and increases stability.

Table 1-8 Advantages and Disadvantages of span wise bending

| Advantages                                                                                                                    | Disadvantages                                                                                           |
|-------------------------------------------------------------------------------------------------------------------------------|---------------------------------------------------------------------------------------------------------|
| i) High manoeuvrability in any speed<br>ii) Reduction in stall speed<br>iii) Access to various L/D ratios by folding the span | i) Increase in weight<br>ii) Smart or appropriate required<br>iii) Threat of improper working mechanism |

### **1.8 NACA Series**

NACA series is an airfoil profile for aircraft wings, NACA (National Advisory Committee for Aeronautics) developed the numbered airfoil system, and then it was further refined by Langley research centre and the united state air force. In the late 1920s and 1930s, NACA developed a set of series and thoroughly tested and designated with a number system; earlier, there was no designation; some random airfoil designs were drawn and experimentally tested for their aerodynamic test results. Langley has developed a point and numbering system for different airfoil cross-sections. A catalogue of 78 airfoils appeared in the NACA report in 1933, and By the numbering system, the engineers could easily differentiate between the shape of airfoils, and numerical assigned specifies camber lines, maximum thickness and special nose features. There are total 5 NACA series as listed below

- i) NACA 4 Digit series
- ii) NACA 5 Digit series
- iii) NACA 6 Digit series
- iv) NACA 7 Digit series
- v) NACA 8 Digit series

Andrew Ellis *et al.* [12] conducted a single actuator mechanism study on an airfoil for morphing its profile, and the proposed mechanism has a linkage array that actuates multiple points inside the airfoil shape. This type of machine consists of multiple components like SAL (shape adaption linkage) and connector linkage and is made of PCC (passive camber connector and the mentioned technique was applied across a number of NACA airfoils for camber morphing mechanism to showcase the mechanism NACA 23012, NACA 23010 and 23015 airfoils large angle of attacks are chosen here to demonstrate the range of possible motion by this type of mechanism, In this array was made of three components SAL, PCC, ACC, the mechanism was tested successfully in the above-mentioned airfoils, and it has been noted that these type of morphing mechanism can be used for practical application.

Michaud *et al.* [13] conducted a study on structural design optimization on an aircraft morphing wing; a load-bearing skin morphing concept to improve performance is introduced to the aircraft wings; this new method uses composite material morphing coupled to wing structure made of aluminium, A series of a validation test is performed to check the structural integrity of the morphing wing. A three-step optimization was developed to reduce the divergence of skin, and the design shows a 56% per cent improvement in performance and an overall total 19% reduction in weight. The experimental outcome for the deformation of the skin under different load conditions has a correlation with FEM model predictions and possesses good morphing capability.

Fincham *et al.* [14] conducted a study on morphing aerofoil's a multi-objective optimization has been incorporated to actuate morphing mechanism into optimization process, the airfoil designed with the morphing system, here in this airfoil FishBAC camber morphing concept was used in the morphing system, different optimization technique was coupled together, A Pareto frontier design with optimum performance was produced which differ from the normal conventional approach, this method could only lead to one single optimum design but capable of highly complex morphing mechanisms which could give us the desired results. When the coupled optimizer was applied to the FishBAC, it was found that there was a large improvement in performance; here author concludes that the simplicity in the FishBAC morphing system with the improvement in performance makes it an attractive option for future camber morphing aircraft.

Secanell *et al.* [15] designed a morphing airfoil using a computational fluid dynamics solver with a sequential quadratic programming algorithm for different conditions of flight for an unmanned aerial vehicle; for studying initial aerodynamic performance parameters are noted down for the designed airfoil, and then aerodynamic shape optimization program is used to obtain an airfoil with optimal aerodynamic performance parameters. From the experiment, it is noted that a morphing airfoil must be a thin airfoil and must require two morphing mechanisms; with the use of a morphing airfoil, it has been found that the flight envelope of an aeroplane can be extended or stretched, and mission profile can be made more flexible. Even its manoeuvrability is increased at low speed with the use of a morphing airfoil mechanism.

Barbarino *et al.* [16] conducted a review on morphing wing aircraft technology with both fixed as well as rotor-wing aircraft, mainly focusing on the structural shape-changing concept. With recent advancements in advanced materials, development in actuation technology, modelling techniques, optimization techniques, the technology programs should match with capability and funding to continue further research in morphing mechanism, morphing technology is not ready for flight, yet much improvement and extended study is required, efforts must be taken by the aerospace industry to take the technology seriously and include it in design alternative, It is estimated that it will take another 10-30 years from now to see aircraft soaring the sky just like birds with wings without an active control surface

M H Love *et al.* [17] demonstrated morphing technology through wind tunnel testing; a model of the Lockheed Martin concept aircraft is designed and tested in a transonic dynamic wind tunnel. The aircraft incorporated integrated foldable assemblies and has a Thermo polymer actuator. The model was remotely moved to different actuating positions such that the data could be collected in testing without taking a pause of the wing fold mechanism movement. DARPA sponsored test was conducted on the morphing concept, and the structural and ability of airframe integrated support was verified, the ability to withstand structural load on the structure was verified, and the wind tunnel test confirmed that the morphing aircraft from Lockheed martin the structure could withstand up to 1G steady flight, In order to provide data above 1G and loads acting while manoeuvring on the morphing aircraft require further development, the author concludes that future development in alternative materials may eliminate this complexity.

Dayyani *et al.* [18] conducted an experiment with elastomer coated composite skin for manufacturing of camber morphing airfoil; the inplane strain capability and anisotropic performance of these corrugated panels make them perfect material for morphing wing application. In this experiment, the geometric parameters are optimized to minimize stiffness and overall weight of the skin, FEM code is used for beam elements to optimize geometric parameters of the corrugated panel, and an experiment is conducted to study the corrugated skin performance for aerodynamic and structural loading, The variation in the mass of the optimized skin for various loads are noted down and investigated, the structural corrugated skin helps us to reduce the number of FishBAC stringers which will, in turn, reduce the weight of the structure.

Austin *et al.* [19] did experimental validation on adaptive wings employing flexible structures using transitional actuators. A model is manufactured with a finite element for fixed contour control with internal actuators. model with adaptive ribs was constructed, and FEM analysis was conducted to make sure that it doesn't overstress the structure. Contour control replica was validated by the experimented method; when observed, this method was capable of reducing wing drag and reduction in induced drag was also noticed; this study recommends the development of lightweight dependable actuators to make wing cross-section profile.

Wenjun *et al.* [20] conducted an experiment on cellular structure for designing an airfoil for morphing wing structure, an irregular honeycomb cellular structure with a negative poisons ratio with the capacity to go through overall displacement, numerical analysis is conducted using ANSYS, it was verified that application of honeycomb structure with a negative poisons ratio is capable of structural deformation and have low inplane axial stiffness and high out of plane flexural stiffness and as per the simulation analysis the cellular structure with negative poisons ratio is useful for morphing wing concepts.

Seber *et al.* [21] conducted studies on the NACA4412 camber morphing mechanism. A sliding motion is provided to the semi-open secondary wing structure supported with slant wedge support driven by actuating mechanism. The hinge less control surface is made out of 0.635 mm thick A12024-T3 panels. A detailed FEM model was generated, and structural CFD analyses were carried out; aerodynamic coefficients and pressure distributions over the chord were calculated. The detailed study concludes that this mechanism can be used in both symmetrical as well as in un-symmetrical airfoils. The actuating force is applied to the cut-out rib edges, and the

results depict that shorter cut out ribs undergo large camber variation as compared to larger cut out rib edges and also provide an increase in flexibility. The geometric results also show that the aerodynamic coefficients depend extensively on camber variation.

Fincham *et al.* [22] adopted a new optimization strategy for morphing systems, and an aerofoil with aerodynamic optimization has been manufactured in which fish Bac system is used in the proposed model in which radial basic function interpolation method of optimization is practised in which only 2-d optimization is considered. Results of NACA 0012, Fish BAC system, and perfect morphed system were compared, which shows an increase in performance with an increase in 2- 4 % drag value. Results were shown using frontier graph values. Using aerodynamic and mechanism optimization together, small technical design problems can be solved. It focuses on how a design can be optimized to reach aerodynamic values.

Zhiping Li *et al.* [23] this technique focus on the effect of airfoil thickness on stall characteristics at low Reynolds number. In this proposed work, GA (W)-1 is taken for the research study, and lift characteristics of different airfoils are collected through computational simulations. On the basis of topological invariant rules cusp, a catastrophic model is introduced to study the effect of airfoil thickness on stall behaviour. Overall lift characteristics are studied and observed that the overall lift coefficient changes as the angle of attack increases or decreases, the difference in the development of separation leads to the difference between the stall to long separation bubble and the airfoil stall changes from a bistable system to monostable system. The method presented in this research is applicable to GA (W)-1 airfoils such as NACA 00 series; it has the potential to apply to other airfoil series.

Shailesh *et al.* [24] studied the effect of Reynolds number on NACA 0012 airfoils for the various angle of attack experimentally and numerically; the study was conducted at various Reynolds number chords wise at the various angle of attack, from the study on NACA 0012 airfoil it has been observed that coefficient of lift decreases and coefficient of drag increases as we increase the angle of attack, from the study it was observed that lift to drag ratio of NACA 0012 airfoil losses its performance as we increase Reynolds number. Also, with an increase in the angle of attack, vortex decapitation increases at the leading edge, which results in overall lift loss. This study concludes that the centre of the pressure plot decreases with an increase in the angle of attack.

Lei *et al.* [25] studied trailing edge separation on a symmetrical airfoil at a low Reynolds number analyzing flow separation at the rear location of an SD 8020 airfoil simulated. Laminar separation bubble and unsteady separation bubble, and vortex decapitation are observed. As the angle of attack reached a specific value, two kinds of vortices were observed. At a low Reynolds number, laminar separation occurs on both surfaces of an airfoil at a small angle of attack. As the angle of attack increases, laminar separation bubbles emerge on the upper trailing edge of the airfoil, and also vortex shedding cycle varies with the change in angle of attack. The intensity of the primary vortex is weaker and imposes less effect on the upper surface pressure distribution, while the intensity of the secondary vortex is stronger, which has a greater impact on the upper surface pressure distribution

Yang *et al.* [26] A theoretical model was introduced at first to evaluate the unsteady heat transfer. In the experiments, a high-speed infrared thermal imaging system was used to achieve temporally resolved measurements of the surface temperature distribution over the ice accreting air foil surface. The transient behaviours' of droplets impingement, dynamic phase and surface water runback changing processes were characterized quantitatively based on the quantitative surface temperature measurements. Based on the time evolution of the measured surface temperature for the rime ice accretion case, the water collection efficiency distribution around the air foil surface was determined quantitatively to estimate the convective heat transfer coefficients over the ice accreting air foil surface

Bae *et al.* [27] conducted a study on a variable span morphing wing, and aerodynamic characteristics of the wing were analyzed; its wing span is varied, and test results were taken; the results show an improvement in aerodynamic characteristics with the use of variable span morphing wing there was a significant reduction in induced drag, which results in improving existing range of the unmanned aerial vehicle, it was found that increase in wing span cause structural deformation, the aero elastic phenomenon must be considered for a variable span morphing wing

Leishman *et al.* [27] conducted an experimental stall characteristic on NACA 23012 at various Reynolds numbers; several miniature pressure transducers and three hot film gauges are situated in the mid of the wing. During the experiment, the airfoil exhibited abrupt stall under static conditions; the dynamic stall was more or similar to characteristics of similar aerofoils. Further

experiments and measurements are required to explain the mechanism and stall characteristics. Positive pitch displacement provides benefits and keeps the boundary layer attached at a very higher angle of attack; this indicates that the static stall characteristics will not be similar to dynamic stall characteristics under testing conditions.

Shiv *et al.* [28] explored the morphing concept by comparing them with bird's wings, and estimation of critical performance parameters was done in which changes in wing area, camber, dihedral, and the effect of drag were compared in gliding flight. The effects of the morphing mechanism on characteristics of drag are studied initially, and it is estimated that thrust vectoring is most effective at low speeds or at high wing loadings; change in fuel consumption, change in weight, fuel fraction are the points to be considered and estimated while implementing morphing wing mechanism.

Bilgen *et al.* [28] conducted a study on variable camber airfoil using a piezoceramic composite actuator; the newly manufactured airfoil employs two active surfaces and a single bar box mechanism as the internal airfoil structure. The selection of boundary conditions allows smooth variation and deformation in the multi-direction from the camber line of the manufactured airfoil; The box mechanism generates preferred reflection and camber alteration as predicted. The wind tunnel experimental results show a good and comparable effect compared with conventional actuation systems. When tested at a flow speed of 15 m/sec, observed an increase in value of lift coefficient, although an increase in drag coefficient was also noted; when results were compared with NACA 0009 airfoil, a 72% increase in the lift was achieved.

Yu *et al.* [29] introduced a new method of kinematic modelling and analysis of six degrees of freedom shape morphing mechanism obtained by a number of rigid sliding panels. This mechanism finds application in aircraft based on parallel robot kinematics. In this method, a passive panel system is introduced, where the panels are joined with links; due to the joining of panels with the links providing three unconstrained degrees of freedom, the newly introduced method will serve as base line methodology for evaluating wing morphing technique using sliding panels. The morphing technique using sliding panel research needs to be extended toward airfoils for further investigation.

In 2007 Bharti *et al.* [30] presented a design that employs planar unit cells of already predetermined shape and layout, changing the wing structure for achieving the desired shape of the wing; this design is practical in morphing in two-dimensional diamond or hexagonal-shaped cells may be used in layouts to achieve wing morphing. The change in shape depends on cell shape. The wing structure is designed using tendon actuated cellular mechanism with a cable actuation mechanism is used; this type of design requires the pin-jointed mechanism; however, the possibility of excessive fatigue and deformation can't be ruled out; using cables for actuation is promising, and the desired actuation can be achieved.

Yousefi *et al.* [31] reviewed and attempted to determine the effective positioning of the blowing and suction slot on the NACA0012 airfoil. The computational studies were conducted with the blowing flow control approach. Downstream tangential blowing improves the lift and drag characteristics. In contrast, perpendicular blowing often has adverse effects in comparison with the desired objectives of flow control results, particularly at angles of attack smaller than stall angle. The effects of suction control flow on the aerodynamic characteristics of the NACA 0012 airfoil were subsequently investigated. In suction, larger amplitude results in a larger effect on the flow field around the airfoil, particularly at high angles of attack. Owing to the use of perpendicular suction, the lift-to-drag ratio increases dramatically and the stall angle improves effectively from  $14^\circ$  to  $22^\circ$ .

In this paper, Yang *et al.* [32] analyzed a five-digit NACA series airfoil by maintaining different Reynolds numbers at different Angle of attack to get to know the maximum lift coefficient value occurring at the various angle of attack for varying Reynolds numbers. In this analysis, the authors used NACA 63018 airfoil, and the Reynolds numbers used  $3 \times 10^6$ ,  $6 \times 10^6$  and  $9 \times 10^6$ . After the result and analysis, they have compared the error of actual values to their analysis values. During the analysis, the static pressure applied on the airfoil for different Reynolds numbers at a different angle of attack is noted; from the analysis part, they concluded that the lift coefficient increase with increasing the angle of attack up to maximum lift coefficient and the drag coefficient increase with increasing of the angle of attack. The maximum Lift coefficient for Reynolds number  $3 \times 10^6$  is at 12 AOA, and for Reynolds, number  $6 \times 10^6$  is at 14 AOA, and for Reynolds, number  $9 \times 10^6$  is at 16 AOA.

Stolzman *et al.* [33] analyzed NACA 4414 airfoil, and it is tested and validated based on experimental and numerical simulation. An attempt has been made to increase the stall angle by incorporating vortex generators. The coordinates of the airfoil are generated using airfoil tools online airfoil. Database the chord length is 100 mm, and it produced 151 coordinates which are then transferred to solid works there the CAD model has been created and was imported to ANSYS 18.2 workbench where the model has meshed in such a way that near the airfoil surface used a fine mesh to capture the boundary layer phenomena and corresponding data used for boundary conditions and model utilized for solving the fluid flow equations are given below. Initially, the airfoil stalls at a 14-degree angle of attack, but after incorporating vortex generators with a certain angle, the stall angle of attack is 16 degrees, and a significant increment in the coefficient of lift is observed.

Fatahian *et al.* [34] conducted an experimental analysis on the wing models generated from NACA 0012 airfoil with different aspect ratios, and based on the results obtained; an optimum aspect ratio is presented with graphs and design values for the validation of the design point. The experimental analysis is critical for any design of aircraft; this gives the exact flow behaviour so that can make necessary changes. Three wing models are created and tested in a wind tunnel whose test section dimensions are 300 x 300mm. The incompressible viscous calculations are used to get the values of lift drag and moment coefficients which are non-dimensional, using the reference area of the wing and moment arm. The velocity of air inside the tunnel is 33.76 m/s, and the lift drag and moment are calculated using the TE81 digital measurement device.

Sokolowski *et al.* [35] aim was to optimize the performance of a NACA 0012 airfoil. This experiment alters the design parameters, such as shape etc., to get an increment in lift and decrement in drag. There is an overall lift-to-down drag ratio increment, which is an important performance criterion for designing the aircraft. This is achieved using the generic algorithm of MATLAB and X-foil; the general NACA 0012 airfoil is obtained through the NACA database, and the coordinates are then uploaded to the X-foil where it is optimized using a generic algorithm and finally generated updated airfoil coordinates which are then used to design and simulate using ANSYS. The generic algorithm gives three design points for optimization; after physical intuition and known airfoil behaviour, we decided to optimize the airfoil at one location.

There is a significant increment of lift and decrement in drag for the modified airfoil using a generic algorithm.

Liu *et al.* [36] conducted a review on the origin of aerodynamic force lift on an airfoil using existing formulae and notations focusing on fluid viscosity. The improvement of lift on a flat plate airfoil is reviewed, and a discussion on physical aspects of the lift coefficient of the flat plate airfoil is discussed. Generation of lift will only happen if the viscous flow is over an airfoil; the aerodynamic force will be generated if it is associated with vortices and velocity fields; hence it can be reinstated that the boundary layer is the direct origin of lift. The following points were summarized. Lift is a result of the pressure difference between the upper and lower surface, which is coupled with boundary layer growth as well as vortices, Lamb vector, and magnitude flux. The lift can only be generated in viscous flow, which exists with viscous drag; the circulation theory, as stated by Kutta Joukowski theorem circulation, is directly related to downwash momentum flux.

Llorente *et al.* [37] conducted wind tunnel experiments on two aerofoil series at high Reynolds numbers; the tested airfoil belongs to wind turbine dedicated families. Comparison between two wind tunnel test result measurements when considering the different test conditions. The Reynolds effect in the range of 3 to 12 million has been investigated, showing an increase in the max coefficient of lift for all airfoils. The experiment shows that the RFOIL for this aerofoil predicts the coefficient of lift maximum at 3 million Reynolds number; the aerofoil was tested and measured with Langley low-turbulence pressure tunnel with the ratio 20:1 for the angle of zero lift the NACA 64-618 were adjusted at -0.4 degree in a smooth case, another NACA 63-615 is corrected at -0.87 with +1 degree wrap around roughness.

Zakaria *et al.* [38] In this work, a study on the influence of airfoil thickness is studied on the location of laminar to turbulent transition points conducted on six different series of airfoils. The series of experiments were conducted on constant Reynolds number and at four different angles of attack. The lift and drag coefficient and transition point have been determined using XFOIL. The experiment shows the in-depth detail analysis of XFOIL for the thickness of aerofoil on the location of laminar, turbulence and transition flow. The investigation was carried out on six different types of aerofoil of the FFA\_W3\_XXX series at a constant Re number, which is  $3 \times 10^6$ , where the maximum L/D ratio obtained by aerofoil thickness was 27% to 35% for

positive AOA when the aerofoil thickness was increased. The obtained values result show the following conclusions, for the upper surface of airfoils, when the angle of attack varies from negative to positive values, the transition point moves towards the leading edge, and for the lower surface of the airfoil, the opposite of the previously observed results is obtained. It has been observed that the increase in airfoil thickness affects the lift coefficient positively and medium thickness provides the best lift to drag coefficient.

Xia *et al.* [39] this paper shows progress in research on the flow structure and aerodynamic characteristics of an airfoil at a low Reynolds number, as encountered by near-space low-speed aircraft and micro-air vehicles, summarized in this review. The designs of various types of laminar separation bubbles and their effects on the aircraft's aerodynamic characteristics are discussed. The receptive, primary instability, secondary instability, and break-down stages of the separation bubble's transition process are all described in detail. Studying an aircraft's aerodynamic characteristics and flow structure at a low Reynolds number is critical for engineering and academic purposes. Flow with a small Reynolds number has been extensively studied as a branch of aerodynamics. Based on the experimental and theoretical design, they summarized the problems of an airfoil at this number and concluded that they are closely related to separation and transition. At low Reynolds numbers, a series of experiments and numerical simulations in the literature capture the structure of various LSBs. Small disturbances develop during the transition, resulting in a series of dynamic evolutions of vortices rolled up by the shear layer in the LSB.

Seifert *et al* [39] The experiment was performed using a NACA 0012 airfoil and a NACA 0015 airfoil at high Reynolds number incompressible flow conditions. The effects of Mach and Reynolds number effects were identified, using the capabilities of a cryogenic-pressurized facility to maintain one parameter fixed and change the other. At low Mach numbers, where high-lift for take-off, landing, or loiter is required, the delay of boundary layer separation allows increased loading of a multi-element high-lift system. According to researchers, periodic excitation introduced into a separating boundary layer upstream of the average separation location can effectively delay it. Oscillatory addition of momentum is two orders of magnitude more efficient than tangential steady blowing for separation control. The method's efficiency was not affected by the forced transition, thickened turbulent boundary layers, or increased Reynolds

numbers. Two dimensional, wall tangential oscillatory blowing is currently used since the magnitude, the frequency, and the average mass flux are conveniently controllable. At the same time, the airfoil geometry is almost unchanged due to the narrow blowing slots. The effects of suction through places and a porous surface on the performance of a supercritical airfoil at off-design conditions were tested experimentally. Suction through a single slot effectively maintained the upstream shock position, re-establishing the pressure recovery and positive  $C_{p,TE}$ , and increasing post-stall lift. The double slot and perforated plate (8% porosity over 7.5% chord) were less effective but more effective in reducing drag. Additional aspects such as device friction, device drag, and flow unsteadiness were not investigated. Strong Reynolds number effects were identified in the airfoils' baseline performance at moderately compressible flow conditions and post-stall angles of attack. The main application of the present control method to these flow conditions is a reduction in the number of elements and their complexity in high-lift systems while maintaining performance and steady flow. In transonic flow, the method could be used to lower the buffet rather than generate a higher lift. Using proper instrumentation and control strategy, closed-loop control could be applied to minimize  $C_p$ , form drag, or moment excursions.  $C_L$ ,  $C_d$ , and wake unsteadiness were found to be highly sensitive to  $C_{nu}$ . It could be safely stated that whenever controlled excitation was applied close enough to the separation location, it proved beneficial, regardless of the Mach number.

Yarusevych *et al* [40] Experimental research was conducted on the boundary-layer separation and wake structure of a NACA 0025 airfoil, as well as the effect of external excitations in the presence of structural vibrations on airfoil performance. Hot-wire measurements and complementary surface flow visualization were used in wind tunnel experiments for three Reynolds numbers and three angles of attack. The findings suggest that acoustic waves are the primary source of energy for the airfoil's separated shear layer. Low-frequency structural vibrations have been discovered to have a significant impact on the performance of airfoils. Some experimental results suggest that the optimum effect occurs when the excitation frequency matches the instability frequency of the separated shear layer at angles of attack below and near the stall angle. The separated shear layer may reattach to the airfoil surface at higher Reynolds numbers, resulting in the formation of a separation bubble. The complex physical mechanism that is responsible for this is still unknown. The goal of this study is to see how external acoustic excitation affects the boundary-layer separation and wake structure of a

NACA0025 symmetrical airfoil at different Reynolds numbers and angles of attack. For all Reynolds numbers and angles of attack examined, boundary-layer separation occurs on the upper surface of the NACA 0025 airfoil.  $Re = 57 \cdot 10^3$  forms a laminar separation bubble when the separated boundary layer re-attaches to the airfoil surface. However, the separated shear layer fails to reattach to the airfoil surface in  $Re = 57 \cdot 10^3$  and  $Re_c = 100 \cdot 10^3$ . Vortex shedding occurs at all angles of attack, with a Strouhal number that is comparable to vortex shedding in the wake of a circular cylinder. The fidelity of these structures is highly dependent on the behaviour of the boundary layer and the Reynolds number. The best effect is achieved at an optimum frequency, which rises with the Reynolds number or angle of attack. Furthermore, the magnitude of the excitation effect on the wake structure is proportional to the improvement in airfoil performance, i.e., an increase in lift and/or a reduction in drag.

Jacob *et al.* [41] During approach flights, fan broadband noise appears to be a major sound source for modern aircraft when the engine power and jet noise are reduced. There are two types of noise sources: self-noise and interaction-noise. Because the azimuthal velocity is much lower at the hub, perturbations generated at the hub are aerodynamically important but not acoustically. Self-noise in the stator exists in theory, but it is outweighed by interaction noise. The position of the vortex detachment moves downstream as the gap size increases and the vortex circulation increases. Several studies have attempted to reduce tip clearance noise using active and passive devices such as novel tip designs. The study compares and contrasts tip clearance self-noise with trailing edge noise. The tip clearance flow interaction with a downstream airfoil is automatically excluded when the study is limited to a single airfoil. Although the tip clearance gaps and Mach numbers considered are not identical to those of a real engine fan, the underlying physical mechanisms of tip vortex formation in the gap region and sound radiation are. In the anechoic wind tunnel of the Ecole Centrale de Lyon, a strong tip clearance flow was obtained in a non-rotating, low Mach number single airfoil located in the potential core of an open jet flow. Two components of tip leakage broadband self-noise were identified in the results. The sound generation mechanisms of the tip clearance self-induced perturbations are difficult to model due to their 3D nature. The research is a first step toward gaining a better understanding of tip clearance flows and the sound produced by them. The airfoil wall and the outer flow produce two highly turbulent shear layers, one near the wall and the other near the blade's base. The major features of the flow that are candidate sources of mechanisms for generating sound in a

turbo engine fan have been identified in this study. Although possible interaction noise sources due to tip clearance flow ingestion of disturbances could be added, these were not found to be significant in the current study with a quiet inflow.

Phil *et al.* [42] Experimentally measured and numerically predicted variations of free-stream turbulence intensity, surface roughness, exit Mach number, and airfoil camber are considered when no film cooling is used on the symmetric airfoils and cambered vanes. The findings presented here are the result of a series of investigations conducted over the past 32 years. Aerodynamic losses on a local and global scale are also calculated. An overall mass-averaged magnitude of energy destruction is determined for each experimental arrangement. When film cooling is used on symmetric airfoils, two different film cooling hole configurations are located just downstream of the passage throat when film cooling is used. Turbulence intensity, surface roughness, blade row interactions, and blade-vane geometry all contribute to turbine blade and vane aerodynamic losses. Variations in Mach numbers, airfoil camber, and film-cooling are also important. A number of recent studies have looked at the aerodynamic losses downstream of subsonic turbine airfoils that do not have any film cooling. Boundary layers, shock waves, and wake mixing all contribute to overall losses, which are proportional to the Mach number. When they quantify heat transfer and aerodynamic performance characteristics of turbine airfoils with different surface finish treatments, tumbling and polishing reduce the average roughness size and improve overall performance. According to researchers, thicker boundary layers that develop over rough surfaces are more sensitive to changes in free-stream turbulence than thinner boundary layers that develop over smooth surfaces, according to researchers. Through a change in the pressure gradient imposed on the boundary layer, the exit Mach number has an effect on the losses. To keep the temperatures of turbine hot-section components within acceptable ranges and temperature gradients, film cooling is used. Film injection from a single row of holes placed on the suction or pressure surface can increase or decrease aerodynamic losses in incompressible flow. Because of shock/boundary-layer interactions, film cooling at the leading edge and early pressure surface region can actually improve aerodynamic efficiency. Losses downstream of transonic airfoils with trailing edge ejection are also being investigated. The leading edge, suction surface, pressure surface, and trailing edge all have holes for film cooling. The largest loss increases relative to the flow with no film cooling as the mass flow rate ratio changes due to injection from holes near the passage throat on the suction surface and holes on the leading edge.

As a result of higher advection speeds and increased diffusion within the wake, the wake downstream of the vane becomes wider at higher exit Mach numbers. Increased intensity levels also broaden wake profiles, especially on the suction side. This behaviour is caused by varying amounts of mixing just downstream of the airfoils, as well as varying turbulent diffusion of stream wise momentum in a direction normal to the airfoil symmetry plane, as evidenced by differing aerodynamic loss profiles magnitudes and levels of symmetry. As the exit Mach number rises, the magnitudes of the area-averaged loss coefficients  $Y_A$  rise as well. The magnitudes of IAL also increase slightly as the level of inlet turbulence intensity rises. The effects of film cooling on hole orientation, shape, and a number of rows are considered for four different hole configurations when suction-side gill region film cooling is added to cambered vanes. As the blowing ratio or the number of rows of holes increases, the magnitude of integrated aerodynamic loss (IAL) also increases. The airfoil configuration, film cooling configuration, and flow Mach number all influence the performance of each hole type. When compared to a smooth blade with no film injection, the magnification of a cambered vane with film cooling increases by 4 to 45 per cent.

Smith *et al.* [42] The retreating blade flow characteristics at high advance ratios have historically limited the maximum flight speeds of traditional rotorcraft. The conventional flight envelope is being expanded to include higher advance ratios above 0.5 due to modern concepts like the X2 Technology TM demonstrator. When reverse-flow angles of attack are taken into account, modelling of the airfoil trailing edge becomes more sensitive. For a turbine operating in a relatively modest 30 degree yawed flow, the influence of the turbine wake results in angles of attack exceeding 30 degrees as far outboard as the 50 per cent radial station. A quantitative analysis of CFD's ability to recreate detailed unsteady physics at all angles of attack is the focus of this research. An O-grid is created to evaluate the NACA 0012 topology, which includes a finite trailing edge. The SC1095 airfoil study used grid sizes and spacing based on the symmetric grid study. For both the NACA 0012 and SC1095 airfoils, extensive computational fluid dynamics studies of airfoil performance were conducted over a range of angles of attack from 0 to 180 degrees. At low-to-moderate angles of attack, some results are provided to demonstrate the validity of the current simulations to experimental and computational data. The behaviour of an airfoil in the deep stall regime, particularly near  $\alpha = 90$  degrees, is similar to that of a flat plate, regardless of camber, according to the findings. The lift of a NACA 0012 simulated with

an O-grid at  $\alpha = 170$  degrees was found to be over 25% greater than the C-grid prediction for the same flow conditions. Experimental validation indicates that the CFD predictions have demonstrated the ability to capture pre-stall behaviour within experimental limits, indicating that the numerical simulations can be valued.

Madhan *et al.* [43] Analysis was carried out on two aerofoils, and their characteristics were studied by varying their camber. The aerofoils used were a seven-digit aerofoil and a supercritical aerofoil, namely NACA 747A315 and NACA SC(2)-0714, respectively. The analysis was carried out with the help of ANSYS FLUENT by varying the values for the angle of attack. A modified supercritical aerofoil was used for this study. The results showed that the varying camber method yielded variations in the generation of lift, and subsequently, the value of drag was also affected. A change in the position of camber in the aerofoil's lower portion exhibited increased values for lift by precisely 20%. This was the scenario observed for the supercritical aerofoil. Comparisons were drawn for the same concerning the seven-digit aerofoil as well. It was found that increased values of lift coefficient were the result (around 12.5%). The interesting point was that even the value of the drag coefficient was altered (reduction of around 9.51%).

Paramasivan *et al.* [44] The motive of this paper was to study the lift and drag characteristics of certain supercritical aerofoils. The analysis was carried out with the help of ICEM-CFD. By varying the values for the angle of attack, comparisons were drawn for the lift and drag coefficient values. Furthermore, the pressure distribution was also studied. The effective nature of supercritical aerofoil forms the crux of this study. Experimental analysis was carried out to compare the results with those obtained post CFD analysis effectively. These aerofoils exhibited a comparatively larger leading-edge radius and decreased curvature mid-span. After comparing the data, it was concluded that the aerofoil 0406 exhibited comparatively greater lift values while generating less drag.

Jianlong *et al.* [45] This paper dwells on wing turbine blade optimality and delves into the question of whether aerofoil concavity can be instrumental in enhancing the aerodynamic performance of wind turbine blades and probes into related topics. Aerofoil concavity, in general, aids in inhibiting flow separation. Experiments were carried out, and it was observed that aerofoil concavity indeed enhanced aerodynamic efficiency by 3-15%. The associated sound

levels were also analyzed, which yielded the result that sound pressure levels showed reduced values of around 9.6-15.8%. Furthermore, vibration analysis was carried out for the blade, and it was found that vibration stiffness exhibited increased values of around 7-4.9%. The blades in question are small horizontal axis wind turbine blades. Even though all these test results do confirm that aerofoil concavity can indeed enhance aerodynamic performance, the large scale implications and advantages require further study.

Lie *et al.* [25] The crux of this paper lies in the flow separation at the trailing edge of the asymmetric aerofoil. This experiment was carried out at low values of Reynolds number. The aerofoil used is SD8020. Vortex shedding from the aerofoil region is being studied along with unsteady bubble bursts. Primary and secondary vortex is also being studied. At regions of lower Reynolds number values, the time dependant CL graph exhibits oscillations for the concerned aerofoil. Both regions of the aerofoil exhibit laminar separation, and the generation of bubbles increases the angle of attack. These are primarily observed at the upper region of the aerofoil's trailing edge. The cycle of vortex shedding is influenced highly by the varying angles of attack. If we go on increasing the angle of attack, there will be a point where these bubbles will undergo separation in an unsteady manner and will generate primary and secondary vortex regions. All these cause the oscillations of aerodynamic characteristics. The secondary vortex tends to have a comparatively stronger effect on pressure distribution of the upper region due to being stronger than the primary vortex.

Sarjito *et al.* [46] NACA 2410 was analyzed after incorporating it with flap and slat. The aerodynamic characteristics were studied for NACA 2410 with and without flaps and slats. Comparisons were drawn in terms of lift and drag values at different values for the angle of attack. CAD models were prepared in AutoCAD and Solid works, while the simulations were carried out with the help of ANSYS 14.5. Pressure distribution and velocity values were also studied along with lift and drag parameters. At higher angles of attack, higher values for lift and drag were observed. This trend showed even higher values for the aerofoil with flap and slat. For the aerofoil without the flap and slat, maximum values for the lift were found at 17 and 55 degrees of angles of attack. And for its counterpart, at a 30-degree deflection of the flap and slat range of -2 degrees, the maximum lift values were observed. Slat, in this case, was at a distance

of 4% from the chord. Changes in flow circulation were also studied. It was studied at different angles for a fixed relative velocity.

### **1.8.2 Selection of NACA 23012 five-digit series**

This work highlights the different types of in-plane and out plane morphing techniques developed in the past decades, mechanisms used, material, and the development and incorporation of these techniques in modern-day aircraft. It extends the research on airfoils with varying the camber along the chord line. Our first challenge was to narrow it down to a particular NACA series since there is a lot of NACA series available; our review of the articles and research led us to NACA 5 digit series, and further, we narrowed it down to NACA 23012 series following observations were noted down for NACA 23012 series.

- i) The general aviation industry has particularly favoured the NACA five-digit airfoils in the United States.
- ii) The maximum camber is placed closer to the leading edge (at  $0.15c$  for the airfoils shown) than was the case for the earlier NACA four-digit airfoils.
- iii) Benefit of this design is a higher  $(C_L)_{max}$  than the four-digit airfoil series.
- iv) Its performance is super-critical and favoured for use in airliner wings for providing high lift with low Transonic drag. it develops a reasonably high lift, a low profile drag and a very small pitching moment coefficient.

And our next step was to design a suitable mechanism to vary the chord length and then manufacture the airfoils with variable camber mechanism testing them in the wind tunnel and Ansys simulation software at different low subsonic speeds and calculating the lift achieved by using the empirical lift formulae and comparing the results observed computationally and experimentally received.

### **1.8.3 Designing variable camber mechanism for NACA 23012**

This mechanism acts when the airfoil is not actuated, and everything will be as in the retracted image. When the mechanism is deployed, the green part and the red part will move back simultaneously. In comparison, the rails will be stiff and attached to the front white rib part. The green joint along with its supporter, will move backwards when the space between the front rib

and back-rib becomes equal to the length of the green joint rib portion. The green part will come up and align with the surface of the whole rib. Till then, it will be moving backwards along with the red portion.

And about the process of how we made it. We made a full rib and split it between about a 25:75 ratio. The white portion is the front rib; the red is the rear rib. Then, the gap in between both as required and made a surface represented as the green portion, creating a uniform profile when fully extended.



Fig 1-16 Variable camber NACA 23012 in retracted mode



Fig 1-17 Variable camber mechanism on NACA 23012 series

After this, we thought of a mechanism where when the green portion will go inside the rib when retracted, and both the front and back ribs will make their own surface. Then we made a channel on which will become the base of movement of green surface attachment providing it directions to move. And a supporter will rail on the channel and will be attached to the green surface at the other end. The channel consists of two plates, one of which is connected to green

surface and another to the channel, and both are connected to each other with the help of sliding mechanisms (sliders, spring pushers/shockers, rollers).



Fig 1-18 Slide mechanism in fully retracted mode

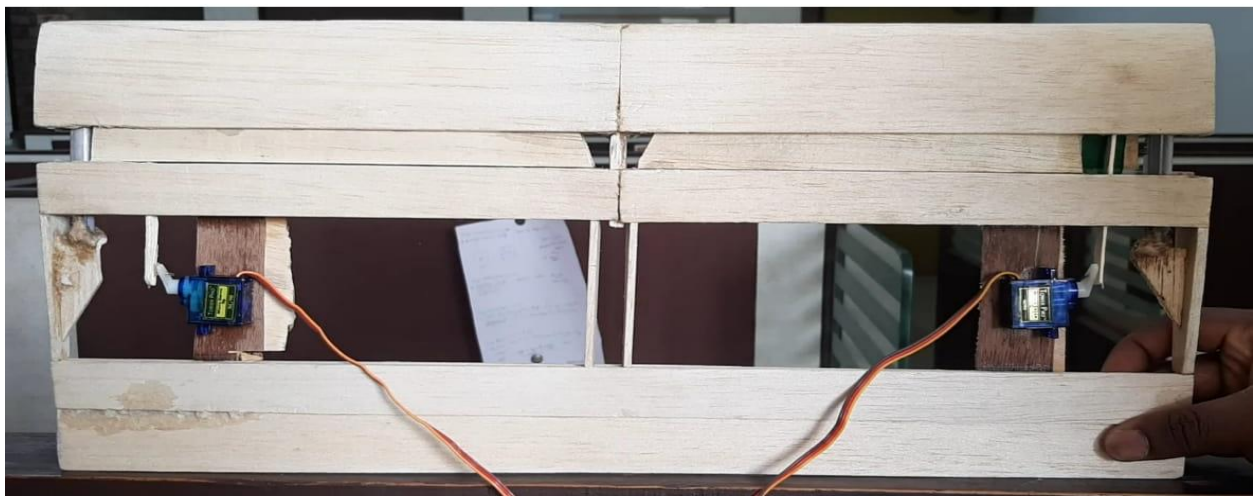


Fig 1-19 Variable camber mechanism fully extended mode

## CHAPTER 2

### RELATED WORKS

## **2. Related works**

### **2.1 Types of morphing techniques**

Researchers in the aviation field are desperately looking for different ways to increase aircraft efficiency by maximizing the Lift to drag ratio and increasing aerodynamic performance. History is often the best answer for future problems, so scientists started looking for the mechanisms in the past and then started solving their concerns with advanced tech. The primary purpose of these researches is to make fixed-wing highly efficient. The primary motivation for morphing is birds, as they use the same wing efficiently in various conditions. Wing morphing is not a new concept. It was in the field from the time of the Wright brothers, and they named it to wing warping as the current aircraft are unable to fulfil the conditions of upcoming problems, so there is a need for change. The wings are built as a compromise geometry that allows a plane to fly in various environments, but performance is always sub-optimal in each. Morphing is a term used in the aerospace industry to describe a group of technologies that improve a vehicle's performance by modifying specific characteristics to better adapt the vehicle's condition to the environment and mission at hand [1]); on the other hand, morphing is short for metamorphosing. Morphing is categorized in three sections as follows

#### **Types of Morphing**

##### **i) In-plane Morphing**

##### **ii) Out plane Morphing**

##### **iii) Airfoil**

#### **2.1.1 Experimental and Computational data analysis of wing morphing techniques**

Moghaddam *et al.* [47] reviewed the recent advances in active and passive flow control techniques. This paper attempts to mention the most technically feasible and market-ready techniques, particularly suction and blowing flow control techniques. The author provides a deep insight into the concept of adverse pressure gradient and its effect on the boundary layer thickness of the airfoil. The techniques such as suction and blowing slots were highlighted in passive flow control. Whereas techniques such as synthetic jets were highlighted in active flow separation control techniques. This study concluded that blowing and suction both result in the increase of lift, but since blowing at the leading edge produces additional vortices causing an

increase in lift, this leads to poor aerodynamic performance. Whereas synthetic jets better stabilize the boundary layer.

Pecora *et al.* [48] The main objective of this paper is to replace the existing flap mechanism with a single slotted flap to achieve morphing of the EASACS25 category flap segment. The rib section is divided into four blocks and is connected with frictionless hinges followed by a hollow shaft brushless motor. The mechanism provides two morphing modes according to the flight settings. Camber mode, where the flap is deployed to get a high lift during takeoff and landing. Morphing mode provides upwards and downwards deflection to the tip of the flap at high-speed flight or cruise conditions. Apart from experimental analysis software model is also analyzed. to get the result values of the mechanism, 3D aerodynamic load distribution over the variable geometry structure is studied in detail with the help of the finite element method. 2D CFD analysis and structural analysis have also been done. The results depict good resistance to weight ratio and load-bearing of the structure followed by the mechanical properties to get the yield/failure value.

Sakarya *et al.* [21] conducted studies on the NACA4412 camber morphing mechanism. A sliding motion is provided to the semi-open secondary wing structure supported with slant wedge support driven by actuating mechanism. The hinge less control surface is made out of 0.635 mm thick A12024-T3 panels. A detailed FEM model was generated, and structural and CFD analyses were carried out. Aerodynamic coefficients and pressure distributions over the chord are calculated. The machine can be used in both symmetrical as well as in un-symmetrical airfoil. The actuating force is applied to the cut-out rib edges. The results depict that shorter cut-out ribs undergo large camber variation compared to larger cut-out rib edges and provide an increase in flexibility. The geometric results also show that the aerodynamic coefficients depend extensively on camber variation.

Santos *et al.* [48] Variable span wing concept has experimented for aerodynamic parameters. The simulation model and prototype of the VSW of a UAV are developed and studied in detail further. An aerospace optimization algorithm is designed for a 2D vortices panel method over a low-speed airfoil SG6042. Various analyses were done over the FEM model of the airfoil, including numerical, flutter, static and structural analyses. The prototype was carefully designed with a monocoque structure and was driven by an electromechanical actuation system. The

composite material was selected. Flight results depict a 22% reduction in drag when the airfoil is subjected to stall speeds of 11m/s-40m/s. A major observation in the reduction of drag is seen at 35m/s over 28.6% and 29.8% Cg of the chord. Max L/D ratio is obtained at 14m/s. All the results were obtained over 30 hours of flight at an altitude of 375m.

Woods *et al.* [49] this study presents a comparison between two different aerodynamic analysis of camber morphing airfoils. This comparison is drawn between the lower fidelity panel method and higher fidelity computational fluid dynamics. A 2D analysis of morphing camber shapes and operating conditions are analyzed. XFOIL and Open FOAM CFD dynamic codes were used to predict the performance of a range of active camber morphing airfoil shapes. The morphing done at different levels at trailing edge causing different magnitudes of displacements reading were noted down at a wide range of angle of attack and following were the findings, overall low fidelity XFOIL panel method has been shown to provide very similar aerodynamic performance when compared with high fidelity Open FOAM computational fluid dynamics code over a wide range operating conditions.

In 2015 Sruthi *et al.* [50] This study aimed to change lift to drag ratio during different flight conditions; in order to achieve it, the wing shape profile must change corresponding to different flight regimes, and the use of smart materials such as shape memory alloy and piezoelectric material increased the function ability of wing. 6 to 8 rib sections covered with latex sheet and insignia cloth material are used to provide efficient movement to wing sections which cause the flow to attach to the surface and delayed separation with that obtained high lift to drag ratio. The test was conducted in a free jet wind tunnel at different Reynolds numbers; through testing, it is found that the variable camber wing produces a higher lift coefficient than the fixed camber wing due to the vibration effect at a quarter of the wing section. Also, at the trailing edge, aerodynamic forces cause to pitch upwards to make flow streamlined; as external skin is applied, camber has been reduced by 2% at the cost of an increase in drag. The wind tunnel results showed significant advantages of the variable camber over the rigid wing in camber configuration, such as higher stall angle and higher lift-to-drag ratio. However, due to the high drag generated by the wing skin of the variable camber wing during baseline configuration, the lift to drag ratio of the variable camber wing was lower than the rigid baseline wing.

Smith *et al.* [51] studied the advanced fighter technology integration(AFTI) aircraft was fitted with smooth variable camber Mission Adaptive Wing(MAW) covered by fibreglass at upper L.E and T.E. There are eight controllable segments in the wing out of 2 are located at the leading edge and six at the trailing edge. A fly-By-Wire control system is used to actuate the segments. Mission adaptive wing showed enhanced low capability and recovered response time during in-flight sweep and manoeuvre. During the test variable camber mission adaptive wings servo actuator mechanisms exceeded 1000-hours service life requirements, no actuator failure or replacements were reported during the testing phase; during the test flight, successful primary and backup control was provided by the mission adaptive wing systems.

Fabio *et al.* [52] The vortex dominated flow around the triple-delta wing aircraft model ADS-NA2-W1 has been investigated by comparing RANS with scale-resolving DDES results and experimental data provided by Airbus D&S. The aim could be a higher understanding of the vortex-dominated flows notably within the sonic speed place order to work out the vary of the suitability of current CFD strategies. With constant side slip angles  $\beta = 50^\circ$  different simulations were conducted, which emphasizes the asymmetry of the turbulent flow, varying the AOA between  $12^\circ < \alpha < 28^\circ$ . The transonic regime of  $M = 0.85$  and  $Re = 12.53 \times 10^6$  has been selected. The RANS turbulence model provides excessive eddy viscosity production in the vortex, with implications on the unburst vortex size, type and velocities. Consequently, the suction peak and the pressure distribution differ from experiments. The secondary vortex is stronger, shifted and hence affecting also the main vortex's position and development. Interesting and promising improvements have been achieved by employing the SA-DDES numerical method. Good results are obtained in the front part of the wing, where only the first vortex is developed.

Guerrero *et al.* [53] Usage of winglets in modern aircraft has proven to be successful in reducing drag, which saves fuel and gives better performance in the field of aviation. This paper mainly gives an insight into the CFD study on winglets used in flights. Generally, traditional winglets are fixed, which gives their best performance against lift induced drag only, which brings up a new innovation in this paper with the idea of using variable angle winglets that gives an extra performance at different points of the flight. Mach number of 0.835 is considered in the CFD simulation. Considering the future prospects, the researchers used winglets at variable angles and plotted graphs of coefficient of drag vs angle of attack and concluded that usage of variable angle

winglets helps decrease drag in terms which help in maximizing the range and payload capacity of aeroplane. It also helps reduce the noise generated by aeroplanes and reduces fuel consumption, which is a major factor in aviation; taking the present global warming and depletion of fossil fuels, it helps in every factor we consider.

Jukes *et al.* [54] compared experimental and numerical analyses of three different wing models used. They have taken NACA 4418 airfoil series used to model the rectangular, tapered and tapered swept back wing. This analysis aims to compare the different aerodynamic parameters like Lift force and Drag force. They have designed the wing in CATIA V5 with a span area of 540mm for rectangular and tapered wings. For the Tapered swept-back wing, the span area is 568.16mm. below-1.26.1. After the comparison between the numerical and experimental analysis, it was consistent with the rectangular and the tapered wings, but for the tapered swept-back wing, the error percentage was 22%. In the Lift performance, the rectangular and the tapered wing was higher than the tapered swept back wing. The tapered swept-back wing was higher than the other two wings for the drag performance.

CHAPTER-3

RESEARCH GAP, RESEARCH OBJECTIVE,  
RESEARCH METHODOLOGY

### **3.1 Scope of Study and Research Gap Identification**

From the literature review, it is found that Lift generated on an aircraft, or a wing, or an aerofoil can be varied with a change in shape profile of the structure to achieve the desired effect; basically, we are varying the camber of the wing there is plenty of lift augmenting devices invented, tested and implemented in aircraft mainly to change the area of the wing. Flaps, slats, slots or a combination of both have been used widely in the majority of aircraft. Major research work has been carried out in changing the camber via external lift augmenting devices; after doing the literature review, it was found an increase of 20 % (approx) lift can be achieved by varying the camber externally, which requires the number of actuators which is basically mechanical linkages which move the control surfaces and can be activated by mechanical, electrical or hydraulic forces since an aircraft there is no compromise on safety, so we have all the above-mentioned systems to move the required lift augmentation device which has a serious drawback that it makes system bulky and increases weight and somewhat complex to operate the available devices.

US air force has shown extreme interest in variable camber aerofoil and wing geometry; a lot of research is done on varying the camber externally on the wing surface, introducing different techniques, and a recent literature survey has revealed NASA is working on shape-shifting wings which is about the varying the geometry of the wing during the flight which will be an advantage at different flight profile of a spacecraft. But few researchers have come up with an idea of varying the camber internally by increasing the area of the aerofoil by a system that actuates and increases the area at different mission profiles whenever required; this not only reduces the weight of the aircraft, it has the potential to eliminate Flaps, and slots which are relied heavily upon by the aircraft for lift augmentation at different flight regimes. This work will develop a aerofoil capable of varying camber internally to produce lift and will be tested at different pressure, velocity and density parameters. The result will be compared with the base NACA profile geometry.

### **3.2 Research Objectives**

- Conceptual design of variable camber aerofoil of 5digit NACA series Samples.
- To estimate aerodynamic and flow performance of NACA 5 series aerofoil samples to validate the effect of camber variation computationally.
- To estimate aerodynamic and flow performance of NACA 5 series aerofoil samples to validate the effect of camber variation experimentally.
- Comparative Validation of computational and experimental test results with respect to the conceptual design.

### **3.3 Research Methodology**

#### Stage 1: Selection of a base NACA Aerofoil section

The first step is to select the NACA aerofoil profile; it has been found in the literature survey that four digits and five-digit series have been widely used for testing and will go for either of the series mentioned above and will be tested for various flow properties and data will be collected. However, a comprehensive literature survey will provide more data on the said aerofoil profile. A 3D printed or balsa wood model will be created for better test results using design software like Creo or Ansys.

#### Stage 2: Design and implementation of internal camber variable mechanism

The main objective is to design and fabricate the variable cambered wing and check its lift and drag at a different angle of attack at different velocities, pressure, and density parameters using experimental and numerical techniques. Compare the lift and drag of variable cambered wing with base NACA profile wing at different angles at different velocities. The primary interest here is to compare the result of two separate profile wings and arrive at a conclusion by analyzing the data.

#### Stage 3: Experimentation of NACA 23012 variable camber mechanism with Subsonic wind tunnel

Various literature surveys have been undergone, and we have come across various experimental methods for performing the experiments. Few of the experimental setups are studied, and from these reviews, we would be selecting the two source-location methods for performing the various experiments in the newly designed variable camber wing. Experimentations would be performed for aligning the experimental results with the results obtained from computational analysis. Calibration of wind tunnel Pressure level and turbulence level inside the wind tunnel will be measured from the experimental method, and after completion of the experiment, comparisons will be drawn between the results and then arrive at a conclusion.

#### Stage 4: Computational data analysis of NACA 23012 variable camber mechanism using Ansys

First, we note down the file's coordinate from the airfoil tools website. We have to use those coordinates in CATIA v5 generative shape design spline from excels, a macro in another window. We have to open the CATIA so that the airfoil gets generated on the XY plane. The chord length of both the airfoil is 1metre; then, by selecting the spline and coordinate of the airfoil, we have to join them, which option is available on CATIA itself. After that, we have to save the airfoil model in CATIA in the format of IGS so that it can recognize by the ANSYS software for further analysis in which we used Ansys fluent. We have to import the file in the geometry of the Ansys, then click on generate for the model to appear on the Ansys; comparative results of NACA 23012 and variable camber airfoil are generated and validated.

# CHAPTER-4

## THEORETICAL DATA ANALYSIS

## **4.Theoretical Data Analysis**

### **4.1 Wind Tunnels**

Wind tunnels help us to experimentally test the interested model under controlled conditions. Wind tunnels can be classified as low speed, high speed and special purpose tunnels. Wind tunnels are designed for specific speed range and purpose, where designers can control the flow and conditions which effects the aircraft by carefully conducting analysis; the engineers can find out the forces on the full-scale aircraft. We can measure the Lift force, drag force, side forces, and pitching moments acting on an aeroplane. Experimental data which is required in aerodynamics is generated with the help of various wind tunnel experiments; Uniform flow is achieved in the wind tunnel test section.

#### **4.1.1 Classification of wind tunnel**

Large tubes characterized by air moving within them are referred to as wind tunnels and are utilized for the replication of the behaviours of a flying object. These are utilized to test scaled models and to predict their interactions with fluid. There are few wind tunnels that are huge in size and can be used to test full-size versions of vehicles. They tend to replicate any object's flight movement by moving air around it.

Strong fans are utilized to create an illusion of flight by moving air through the tube. The model is securely placed in the tunnel with limited movement. The model can be as large as any spacecraft or may even be something as simple as a tennis ball. The movement of air is investigated in multiple ways. Colour or smoke can be thrown into the air, and flight characteristics can be calculated. The addition of threads can also help to demonstrate how the air moves. The force of the air on the item is frequently measured with special devices.

Wind tunnel research has a wide range of applications. Different spots on the model represent or signify air pressure measurements along with variables. These are very crucial as these values provide valuable data regarding the wind load distribution. Aerodynamic research involving these wind tunnels has been very instrumental in tackling design complications and complexities in different sectors- aerospace, railway, civil etc.

## **4.2 Types of wind tunnels**

(a) speed (Mach no.)- Wind tunnels can be categorized as low-speed or high-speed. As the name implies, high-speed wind tunnels operate at speeds when compressibility effects are significant. Subsonic, transonic, supersonic, and hypersonic wind tunnels are also categorized depending on the Mach number. Categorizing high-speed and low-speed wind tunnels- subsonic, transonic, supersonic, hypersonic, and hypervelocity. To replicate temperatures of flight at speeds of 10,000 miles per hour and higher, it is necessary to heat the test air above the melting point of ordinary structural materials; as a result, they operate on an impulse principle and for very short periods. These are normally approximately on the order of a few thousandths of a second.

(b) Operation mode (Pressure storage, in-draft or Pressure vacuum type.) c) Test section type: open, closed, or semi-enclosed.

## **4.3 Instrumentation and calibration of wind tunnels**

Wind tunnel balancing calibration is crucial for avoiding measurement acquisition errors is clear at this stage. For coupling effect reduction and to evaluate the DAQ behaviour, calibration and, if necessary, recalibration are required. As a result, the execution of a static calibration is a primary objective followed by a dynamic calibration.

Although it may appear such at first glance, static balancing calibration is not a simple step in achieving adequate system correctness. Instead of utilizing the wind tunnel, this may be carried out with a separate balance. Calibration is much easier with detachment of this balance from the wind tunnel than with the attachment of balance with the tunnel's mounting supports. The ability to carry out testing for each axis in every direction and in both directions to determine the behaviour of the acquisition system is unquestionably simpler.

This calibration concept assures that the balance's and test chamber's reference axes are the same. In theory, outside of the test room, it has been calibrated and then may be placed accurately after establishing test chamber's local vertical and horizontal directions. Prior to performing a proper wind tunnel balancing calibration, various processes must be completed to determine fixed and trustworthy axes. As for the first step inside the test chamber, the balance must be installed in its

final fixed position. Now inside the test chamber, accuracy in determining the direction of local wind flow is achieved.

The flow direction is determined using a variety of devices, the most popular of which are five-hole, cylindrical probes and so on. Now, in the test chamber, when we place the probe angled with the flow direction, the external probes exhibit a difference in the recorded pressure. Pre-calibration of the curve can be used so that the angle of the flow may be estimated. We can also reorient the probe. The lift values, along with the drag value, are now determined. Lift is perpendicular to the wind flow, whereas the drag force is parallel. The axes of force measurement must be defined correctly. This system is now ready for the calibration phase after ensuring horizontal flow.

#### **4.4 Pressure measurement techniques**

In wind tunnel experimentation, we can measure pressure distribution by various methods, which are explained in detail below

##### **4.4.1 Manometers**

The simplest basic manometers compare the pressure of a gas or liquid to the pressure of the atmosphere. Manometers are available in multiple dimensions. The concept remains the same in terms of the detection of pressure differences. However, there can be improvements in the degree of precision. Manometers require minimal maintenance and are exceptionally precise since they have no moving components. A manometer gauge is a very simple yet incredibly effective instrument for measuring pressure. Traditionally, the end of the tube is left exposed to ambient pressure. The hose is linked to an extra pressure source through a gas-tight seal. A manometer gauge, while most commonly associated with gas pressures, can be utilized for the measurement of pressure exerted by liquids.

Because there are no mechanical parts on the manometer pressure gauge, it requires very little maintenance and is exceptionally precise. As per hydrostatic equilibrium, for a liquid at rest, pressure is equal at all points. The pressure on each end will be equal if the manometer has both ends exposed to the atmosphere. The liquid levels on both ends will be the same. In case we

expose one end to the atmosphere and keep the other end linked to a separate gas/liquid supply, various pressures values will be generated.

When there is a difference in pressure between the fluid supply and the ambient pressure, the measuring liquid is pushed down. As a result, on one side, the level of liquid will be higher than on the other. The side with the liquid under higher pressure will be at a lower level than its counterpart. The movement of a playground see-saw is maybe the clearest way to illustrate this. The higher the value of the weight, the lower will be that side of the see-saw. The most popular device is the classic U-Tube. It is characterized by an end exposed to the atmosphere and the other to an external source. It is feasible to compute the pressure difference with reference to the difference in the height of the liquid on both ends of the manometer.

Both ends of a differential U-tube are filled with fluids having different properties and pressure. Instead of using an outside pressure as a reference, the pressure can be determined directly.

The inverted U-tube design is used to monitor low-pressure between two sites with a high degree of precision. At either end, liquid gets delivered, with a volume of air acting as a separating medium between both. Air may be vented from or entered into it using a tap. As a result, the pressure differential is adjusted, and accurate measurement is obtained. The micro-design is a modified variant of the basic U-tube, with a bigger cross-section on one side. Measurement of pressure difference is almost identical, but a U-structure tube allows it to detect minute pressure variations. An inclined manometer, as the name implies, has a progressive inclination in the design. This enables incredibly precise measurement of tiny pressure. When the manometric characteristics of liquids are comparable, it is employed. It is incredibly efficient and simple to operate because of its simplicity, lack of maintenance, and absence of moving components.

#### **4.4.2 Pitot, Static and Pitot static tubes**

The Pitot tube is used to convert the flow's kinetic energy into potential energy. It is primarily done for the detection of fluid velocity. This conversion is carried out at the stagnation point. This further leads to a higher pressure than the dynamic pressure. This conversion from static to dynamic force is measured using a differential manometer.

A pitot tube is utilized in a broad range of applications involving flow measurements, including racing vehicles and Air Force fighter planes or to monitor airflow in pipes, ducts and so on. These are simple, dependable, affordable, and ideal for a variety of atmospheric conditions, be it high temperature or broad pressure ranges. They do have limitations in terms of range and precision. Pitot tubes are a low-cost alternative to orifice plates. The fundamental property that differentiates it from an orifice is its capability to detect velocity at a specific point, whereas orifice plates are known to measure the flow velocity over the whole flow stream. Another important point is that these pitot tubes can be inserted into pipelines having high-pressure fluids. This is known as hot tapping.

Both the pitot tube and the pitot-static tube are used to determine the fluid's local velocity. They are, however, constructed differently. A single inner tube makes up the pitot tube, which is used to monitor stagnation pressure. Using the differential head, a second hole is placed someplace else to monitor static pressure and therefore discover the flow velocity.

The pitot-static tube (Prandtl tube) is a device that monitors both stagnation and static pressure. It has two holes. The inner one is used to detect stagnation pressure and is hence referred to as a stagnation tube. The outer is used to detect static pressure and is referred to as a static tube. The inner tube is pointed upstream, while the outer tube's entrance is perpendicular to the flow lines.

#### **4.4.3 Three-component balance force measurement**

The force balance is the most basic sort of instrument.

The measurement of the aerodynamic forces acting on the model and its moments is carried out using force balances. On this page, we'll go over a simplified measurement of the lift force in order to teach the fundamentals of force balances. Because it only gauges one force, this setup is known as a one-component balance. Lift, drag, and pitching moment may all be measured using more advanced three-component balances. The 3-Component Balance is a user-friendly support system.

The balance is linked to the vertical wall of the wind tunnel's operational area. It is made out of a mounting plate attachment linked to the operational area. When looking at the balance from the front, air should flow from right to left. A mechanism holds a triangle force plate to the mounting

plate, constraining its movement to a plane parallel to the plane of the mounting plate while allowing its rotation around a horizontal axis. The configuration gives the required 3 DOF.

#### **4.4.4 Six component balance force measurement**

Wind tunnel balances come in multiple dimensions; Some balances are positioned outside the model and hence are referred to as external balances. These are also sometimes positioned within or outside the wind tunnel chamber test section. One major hurdle that these provide is the interference to the wind flow. However, the wind tunnel's capability to alter test models effortlessly, on the other hand, gives the wind tunnel facility a lot of versatility. These balances come in a variety of levels of sophistication in terms of the quantity of measuring channels, which ranges from one to six. Internal balances provide no wind flow interference because the balancing components are positioned inside the model. However, there is a need for mechanical support so that it is well-positioned in the test chamber, and the option of altering the orientation of the model is feasible. Because this balance must be mounted inside, the test model is equivalent to or more complicated than models for scan valve systems. As a result, this option does not allow for the testing of several models. Propellers and other spinning models require rotary balances. External balance has six components.

The commercial balances have six fully linked channels, and the manufacturer provides a 6x6 decoupling matrix. This implies that the measurement of the forces and momentums is carried out with the help of a sophisticated combination of the six-channel signals and a fully decoupled decoupling matrix. The fundamental disadvantage is that recalibration necessitates the use of a specialized calibration facility, which is highly costly.

A full mechanical decoupling of the six load components is the opposing balancing design approach. The output of each channel will be characterized by a unidirectional load signal (traction-compression), with the lift, drag, lateral pressures, and pitching, balance and yawing momentums making up the signal's composition. Three vertical sensors could be used to obtain lift, pitching momentum, and balance momentum. Then to obtain the drag and yawing momentum, two horizontal sensors in the wind flow direction are utilized. In addition to this, one needs to measure the lateral forces as well, which is facilitated by one horizontal sensor

perpendicular to the flow direction. Because this grid is virtually diagonal, calibrating these balances is significantly easier. In actuality, there is a certain coupling effect arising from the absorption of the lateral forces by the sensors. This relationship is especially important when it comes to lift and drag. Balance support, in addition to the model supports, is required to secure the balance of the structure. Because the supports are located outside of the test chamber, the size and the weight are not a major challenge; the only criterion is that they have high rigidity. It's also worth noting that if accurate aerodynamic results are obtained with the help of the external balance, then it is predictable that the same will also be applicable for other models ranging from vehicles, buildings and so on.

#### **4.5 Pressure Plotting curves**

Different types of plots can be plotted with the help of wind tunnel experimental data; with the modern six-component balance, we can plot the Lift Vs AoA curve, Drag Vs AoA curve and different pitching moments and side forces also can be plotted for an airfoil and wing cross-section.

A low-speed air stream in a wind tunnel is characterized by the pressure parameter across the test section, temperature, and flow direction settings in the test chamber of the wind tunnel. To achieve desired test result, special attention must be given to the calibration of the subsonic wind tunnel and to achieve the desired condition inside the test section following points must be noted while operating the wind tunnel. By following the below-mentioned procedure, the effectiveness of the flow velocity and Reynolds number achieved for any specific model testing can be increased.

- i) Calibration of true airspeed inside the test section
- ii) Determining the flow angularity (pitch, yaw and side forces) in the test section
- iii) Flow level turbulence
- iv) Velocity profile across the horizontal and vertical axis of the test section to determine the flow quality in the test section
- v) Wake study and determination of flow across any model

After successful calibration of the wind tunnel, a variable camber airfoil of the NACA 23012 series is placed in the wind tunnel, and results are noted.



Fig 4-1 System with DAQ software for measuring six component forces



Fig 4-2 Subsonic wind tunnel with six-component force measurement set up and DAQ software for plotting the results

## **Aerodynamic Testing on NACA 23012 Airfoil**

An airfoil is a two-dimensional cross-section of a wing sliced in the general direction of the flow. The airfoils display the aerodynamic shape used to produce a pressure imbalance. The net force of the pressure imbalance (in a real fluid, frictional forces are also present), summed over the wing, is resolved into lift and drag. By definition, a lift is the net force component perpendicular to the flow and drag. The lift is the net force component perpendicular to the flow and is the net force component parallel to the flow. Various forces acting on an airfoil can be determined experimentally by mounting a model on a six-component balance and measuring the force directly, it can be determined by integrating a measured static pressure distribution over the entire surface, or it can be determined from a momentum balance on a control volume which contains a model. This momentum balance would require velocity measurements both upstream and downstream from the model. This is the method that will be utilized in this experiment.

### **Procedure:**

1. Check the wind tunnel for any loose parts
2. Adjust the NACA 23012 airfoil in the test section at a  $0^\circ$  angle of attack
3. Set the digital gauges to zero position
4. Run the tunnel at 20 m/sec for the initial set of reading
5. Note down the values of lift and drag from the digital gauge
6. Repeat the experiment at different AoA and different velocities

#### 4.5.1 Wind tunnel Experimental results plotting of coefficient of lift Vs AoA for NACA

**23012**

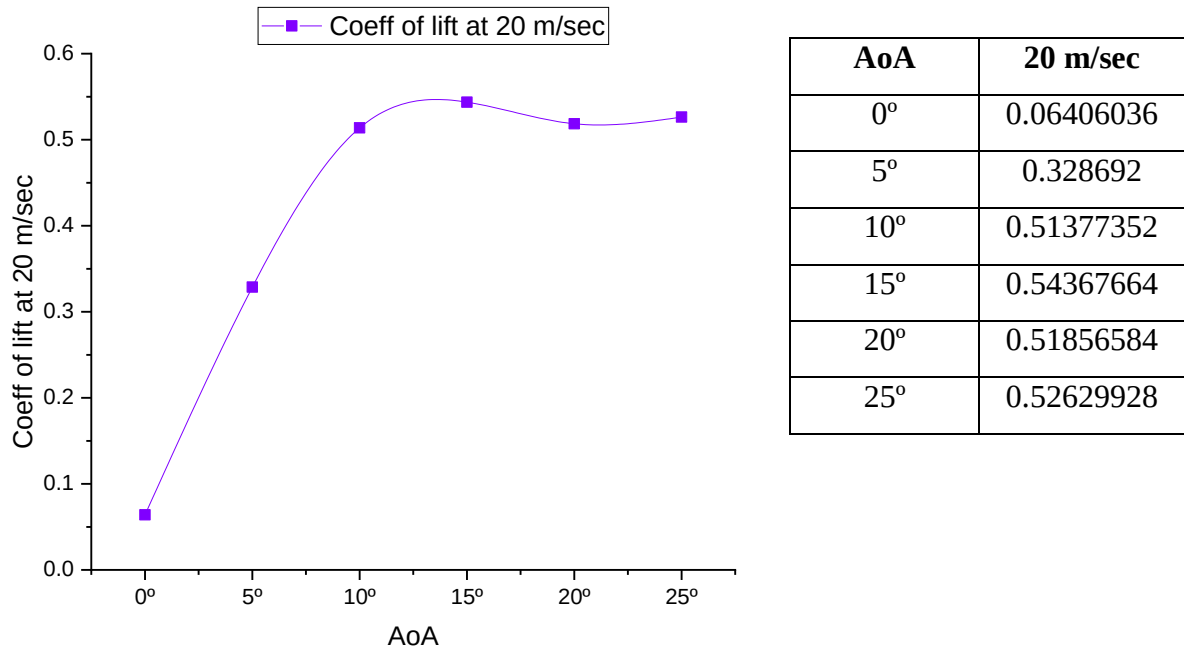


Fig 4-3 Coefficient of lift Vs AoA at 20 m/sec for NACA 23012

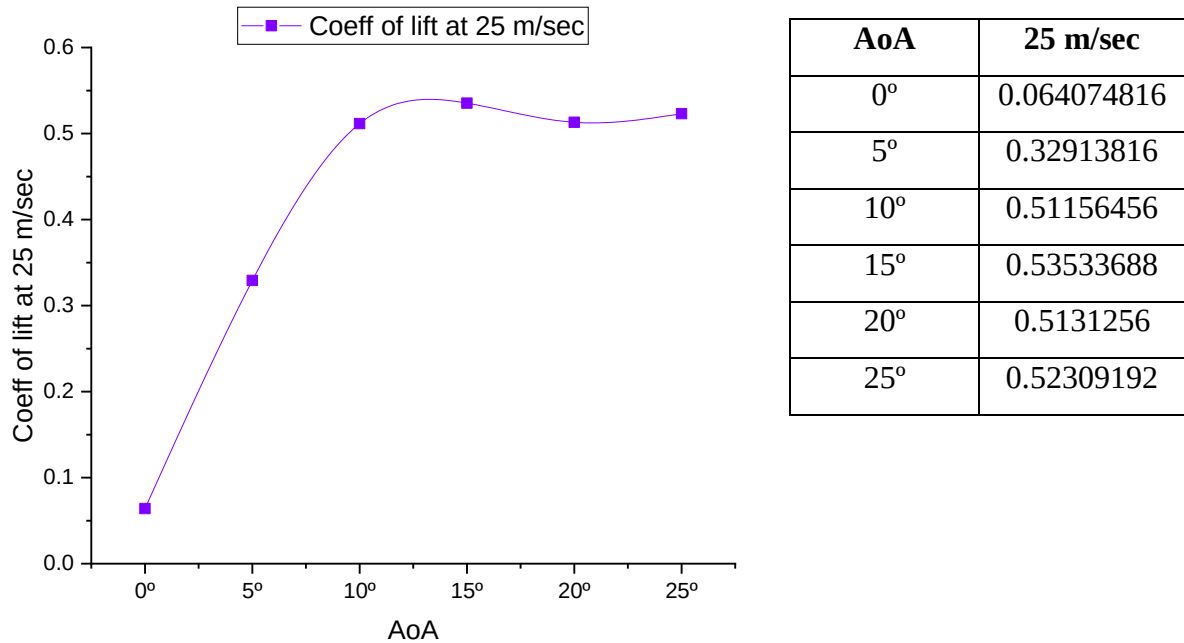


Fig4-4 Coefficient of lift Vs AoA at 25 m/sec for NACA 23012

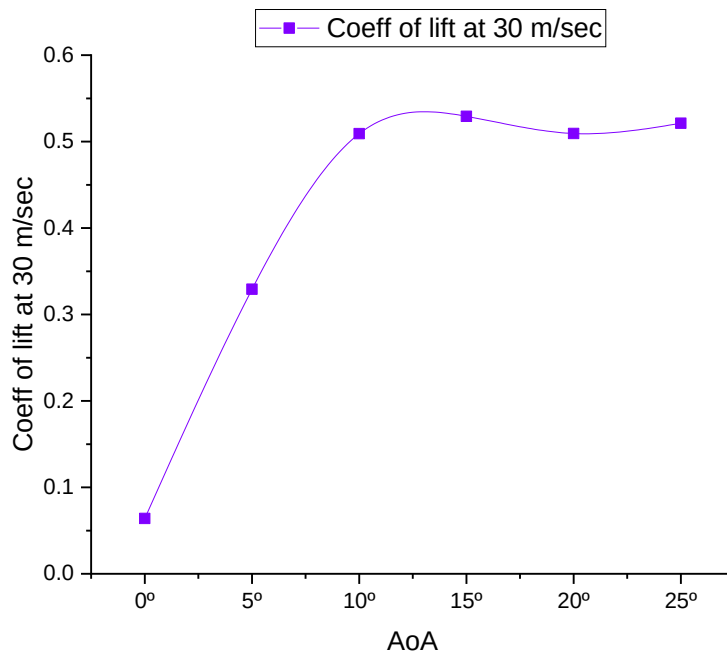


Fig 4-5 Coefficient of lift Vs AoA at 30 m/sec for NACA 23012

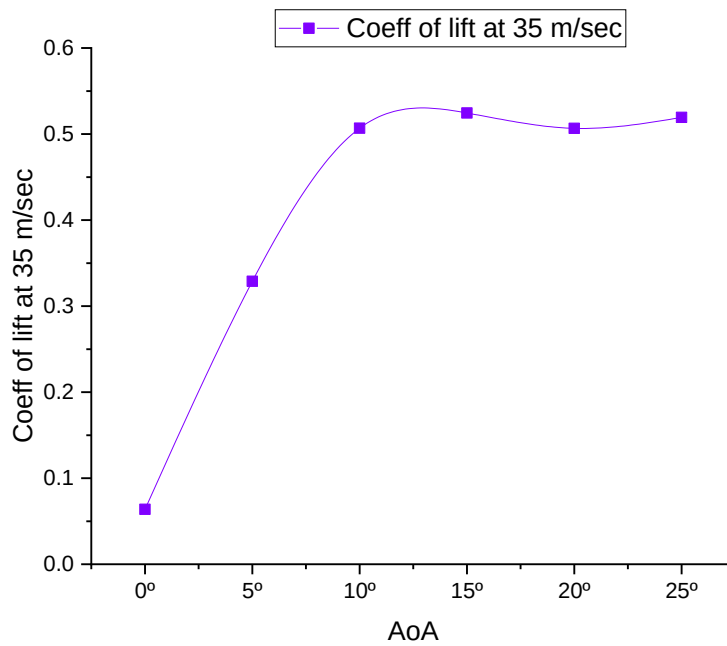


Fig 4-6 Coefficient of lift Vs AoA at 35 m/sec for NACA 23012

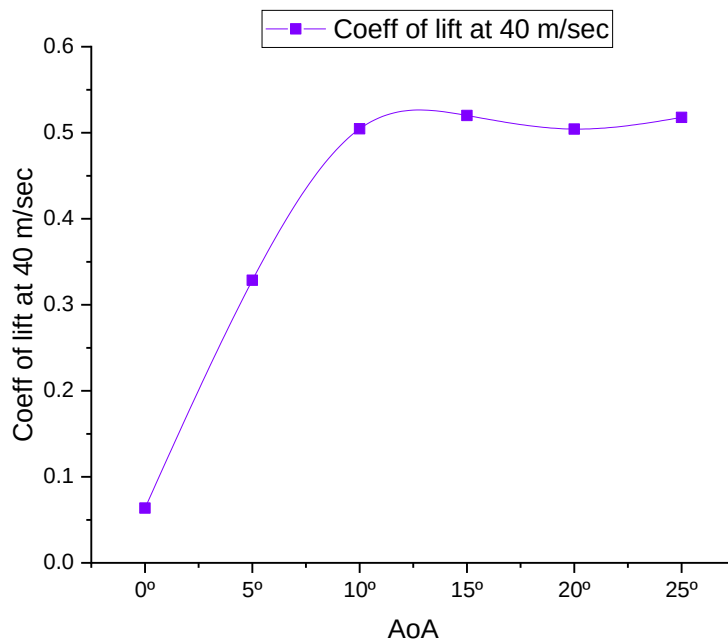


Fig 4-7 Coefficient of lift Vs AoA at 40 m/sec for NACA 23012

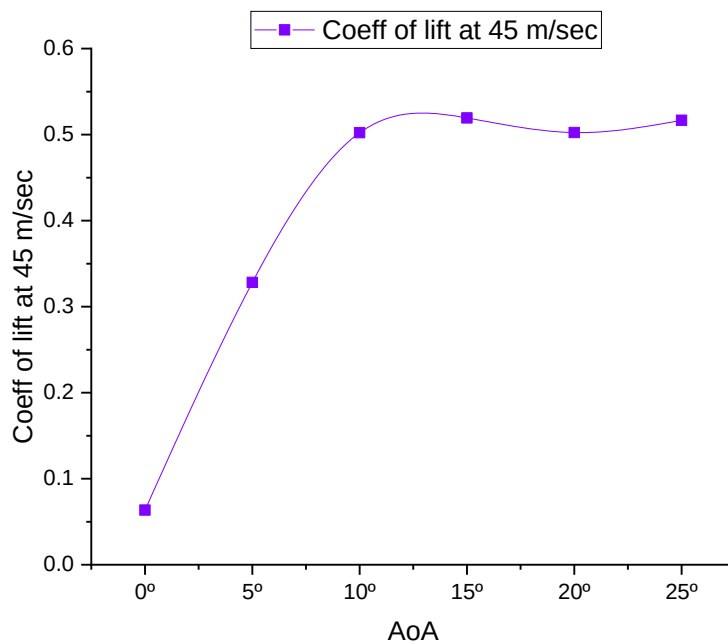


Fig 4-8 Coefficient of lift Vs AoA at 45 m/sec for NACA 23012

#### 4.5.2 Wind tunnel experimental plotting of coefficient of drag for NACA 23012

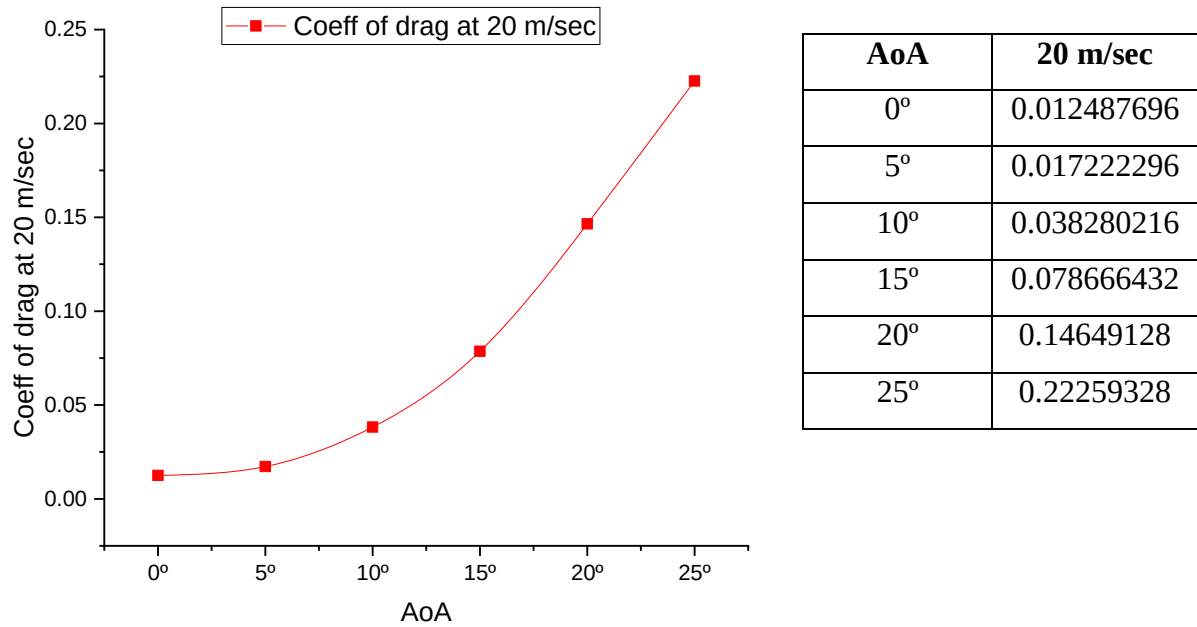


Fig 4-9 Coefficient of Drag Vs AoA at 20 m/sec for NACA 23012

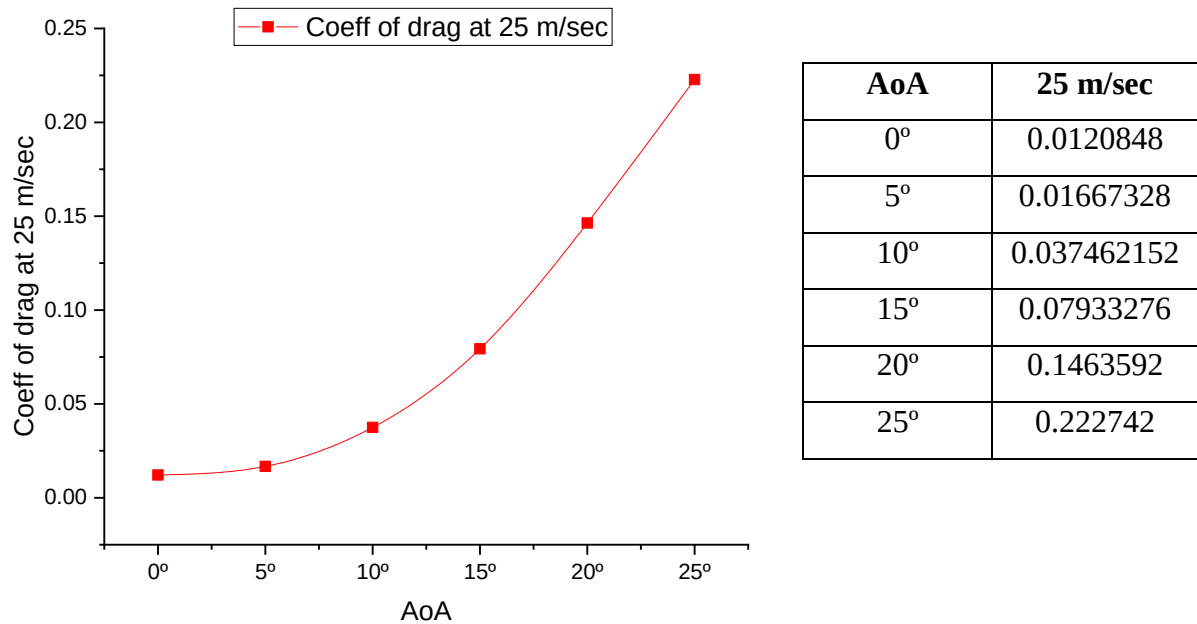


Fig 4-10 Coefficient of Drag Vs AoA at 25 m/sec for NACA 23012

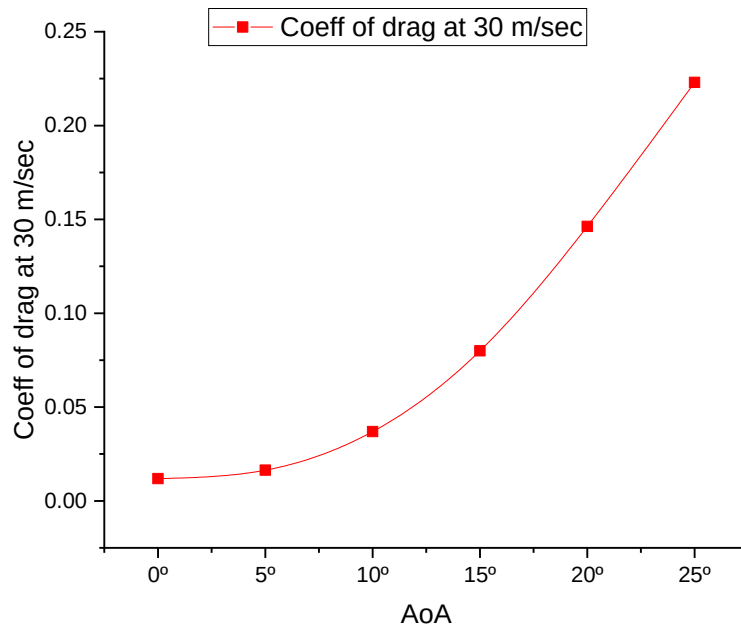


Fig 4-11 Coefficient of drag Vs AoA at 30 m/sec for NACA 23012

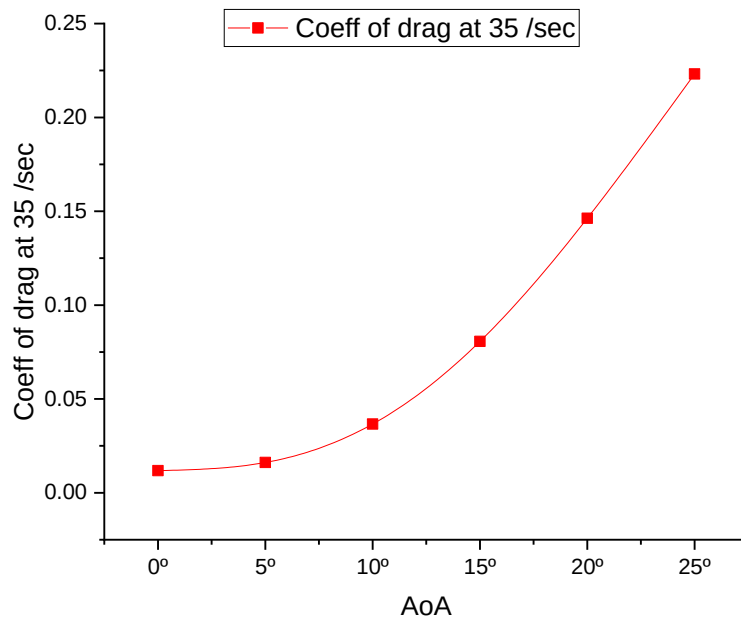


Fig 4-12 Coefficient of drag Vs AoA at 35 m/sec for NACA 23012

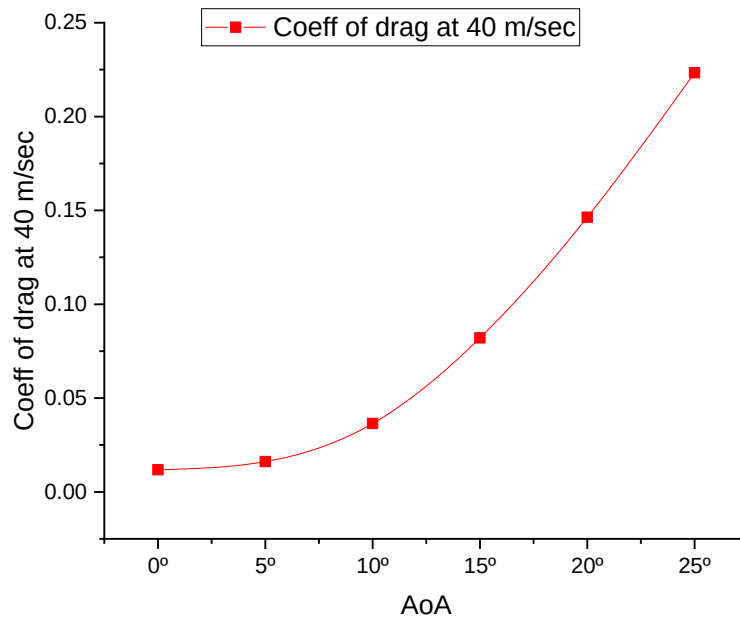


Fig 4-13 Coefficient of drag Vs AoA at 40 m/sec for NACA 23012

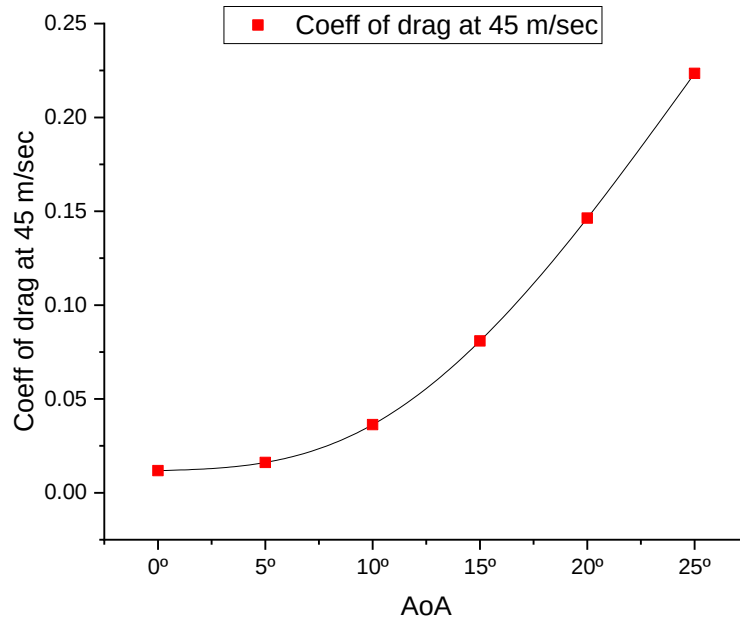


Fig 4-14 Coefficient of drag Vs AoA at 45 m/sec for NACA 23012

#### **4.5.3 Wind tunnel Experimental test results of variable camber NACA 23012**

The same experiment procedures are repeated for the variable camber NACA 23012 airfoil series, and the coefficient of lift and drag is noted down.

**Procedure:**

1. Check the wind tunnel for any loose parts
2. Adjust the variable camber NACA 23012 airfoil in the test section at a  $0^\circ$  angle of attack
3. Set the digital gauges to zero position
4. Run the tunnel at 20 m/sec for the initial set of reading
5. Note down the values of lift and drag from the digital gauge
6. Repeat the experiment at different AoA and different velocities

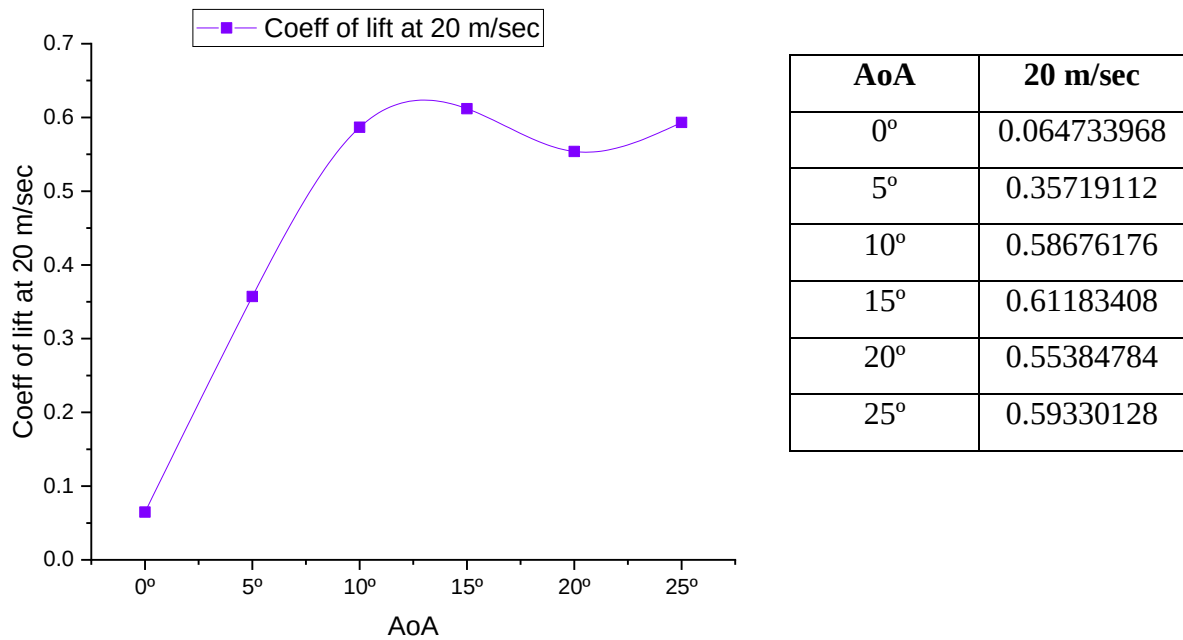


Fig 4-15 (Coefficient of lift Vs AoA at 20 m/sec for Variable camber NACA 23012)

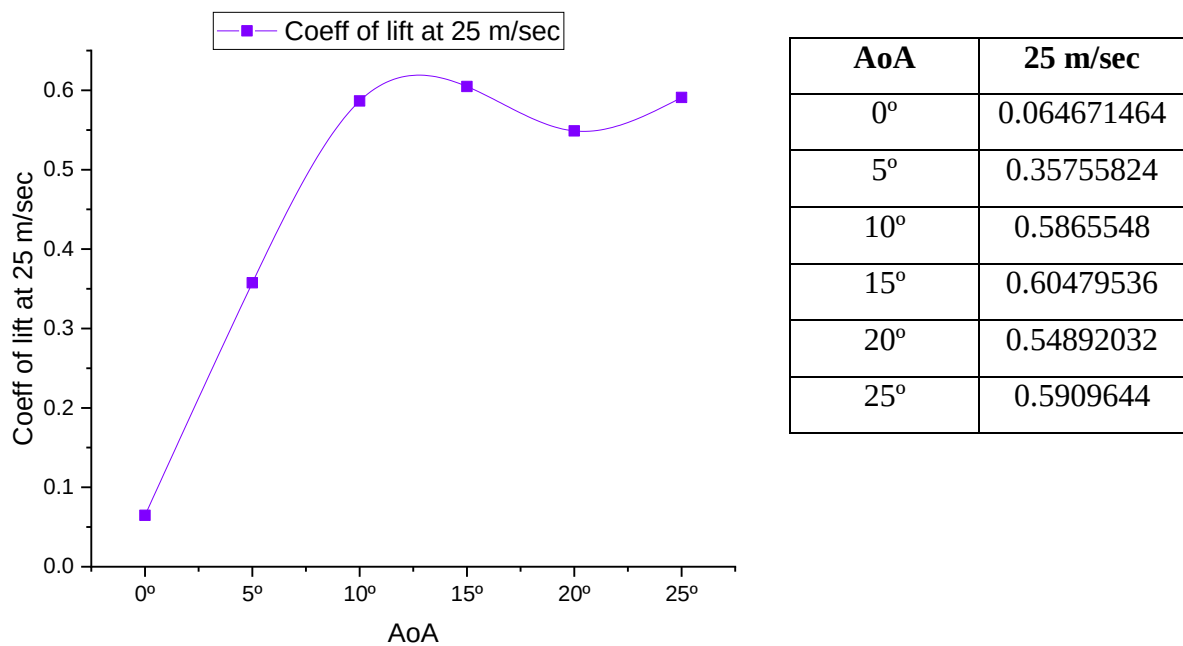


Fig 4-16 Coefficient of lift Vs AoA at 25 m/sec for Variable camber NACA 23012

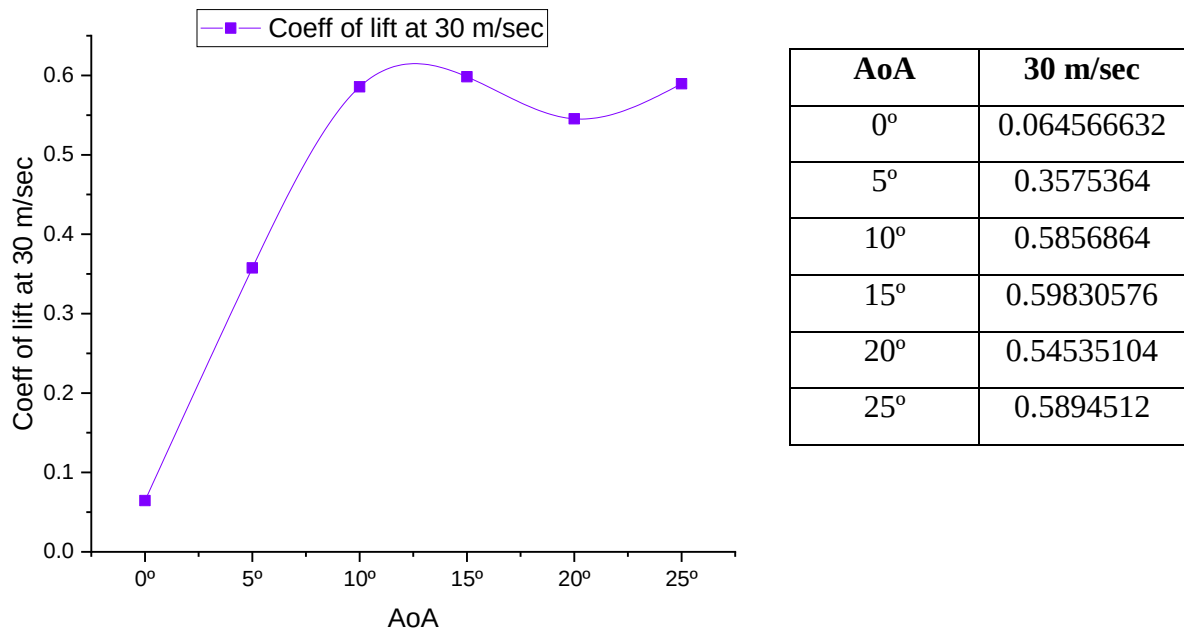


Fig 4-17 Coefficient of lift Vs AoA at 30 m/sec for Variable camber NACA 23012

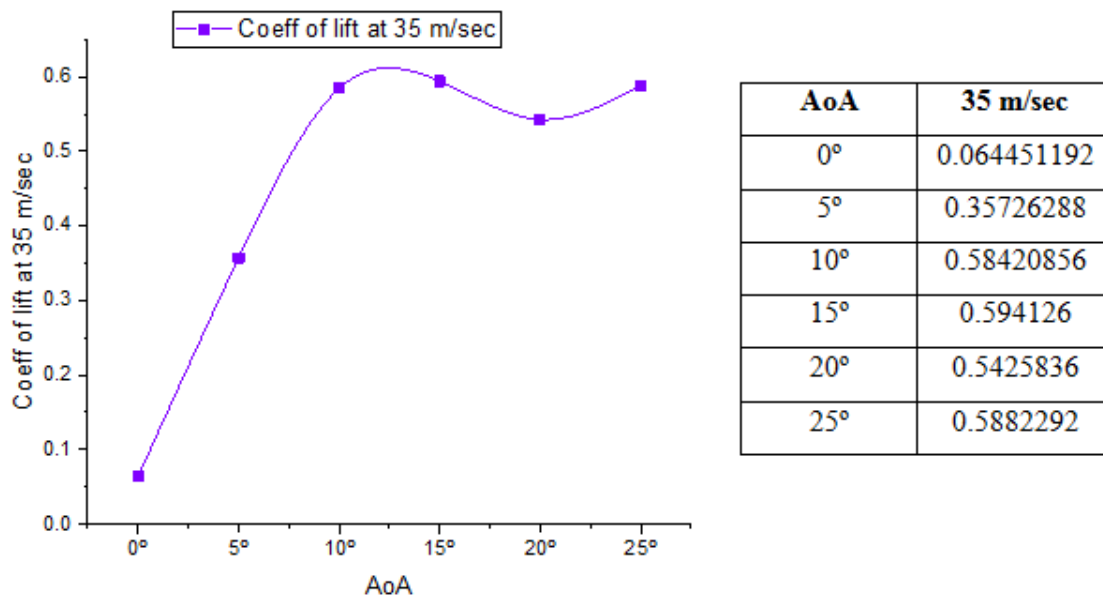
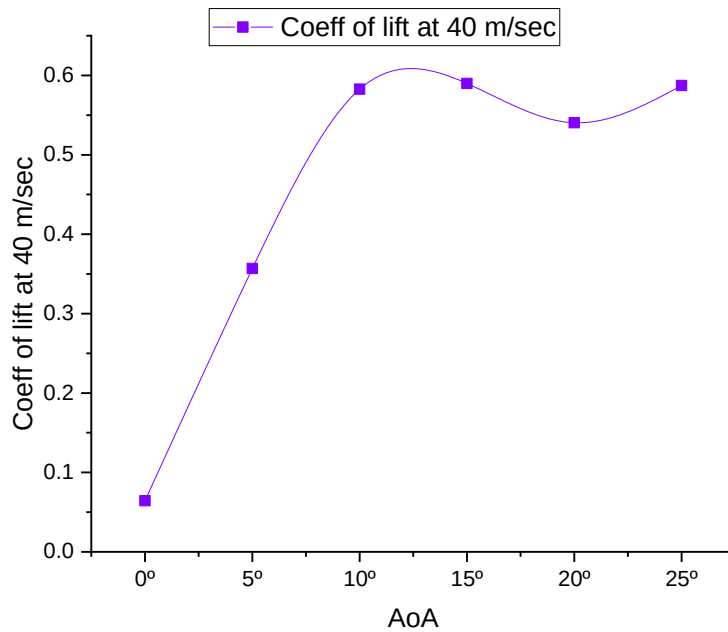
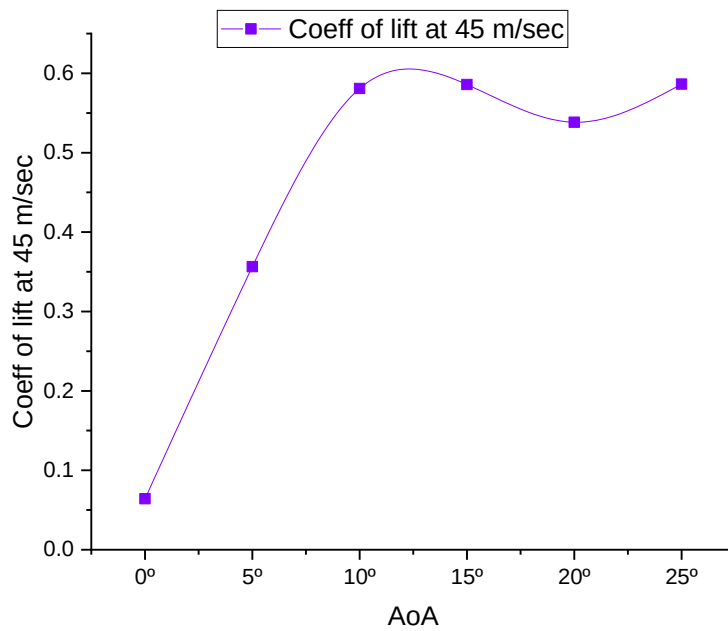


Fig 4-18 Coefficient of lift Vs AoA at 35 m/sec for Variable camber NACA 23012



| AoA | 40 m/sec   |
|-----|------------|
| 0°  | 0.06433752 |
| 5°  | 0.35684688 |
| 10° | 0.58260176 |
| 15° | 0.58990048 |
| 20° | 0.54032056 |
| 25° | 0.5872516  |

Fig 4-19 Coefficient of lift Vs AoA at 40 m/sec for Variable camber NACA 23012



| AoA | 45 m/sec    |
|-----|-------------|
| 0°  | 0.064228112 |
| 5°  | 0.35639552  |
| 10° | 0.58092528  |
| 15° | 0.5857956   |
| 20° | 0.53839552  |
| 25° | 0.58637696  |

Fig 4-20 Coefficient of lift Vs AoA at 45 m/sec for variable camber NACA 23012

### 3.5.4 Wind tunnel plotting of coefficient of drag for variable camber NACA 23012

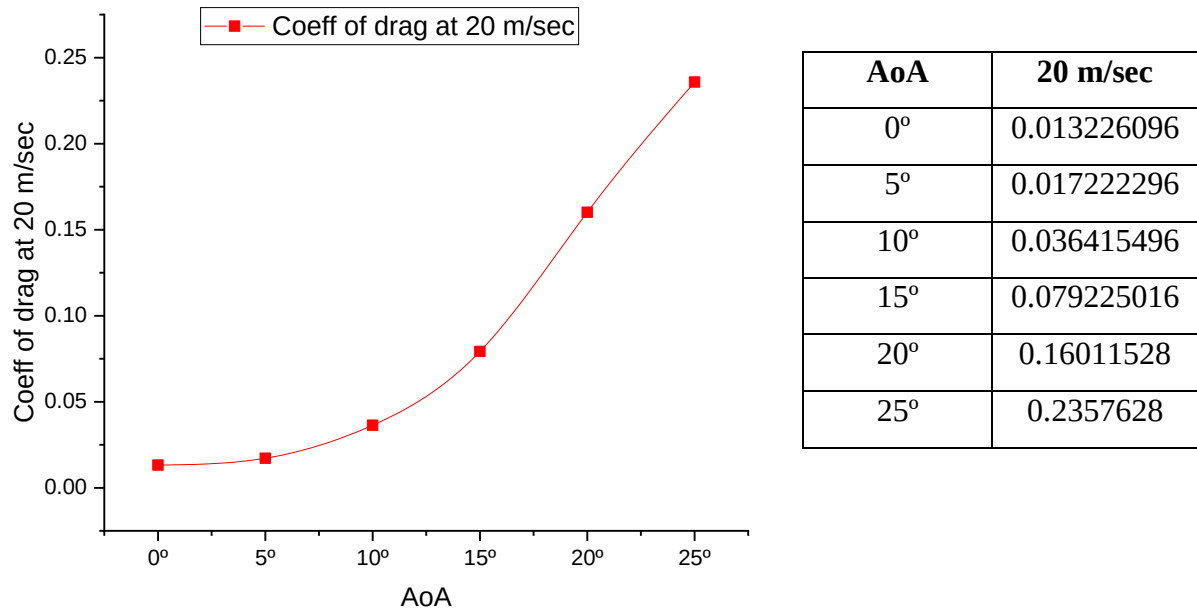


Fig 4-21 Coefficient of drag Vs AoA at 20 m/sec for variable cambered NACA 23012

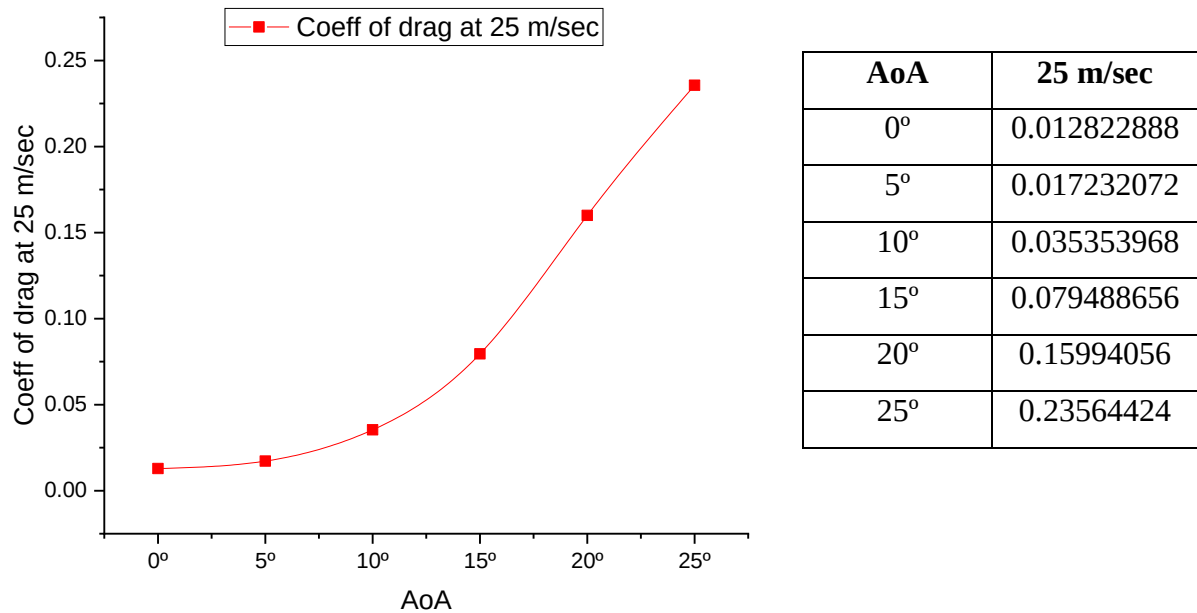


Fig 4-22 Coefficient of drag Vs AoA at 25 m/sec for variable cambered NACA 23012

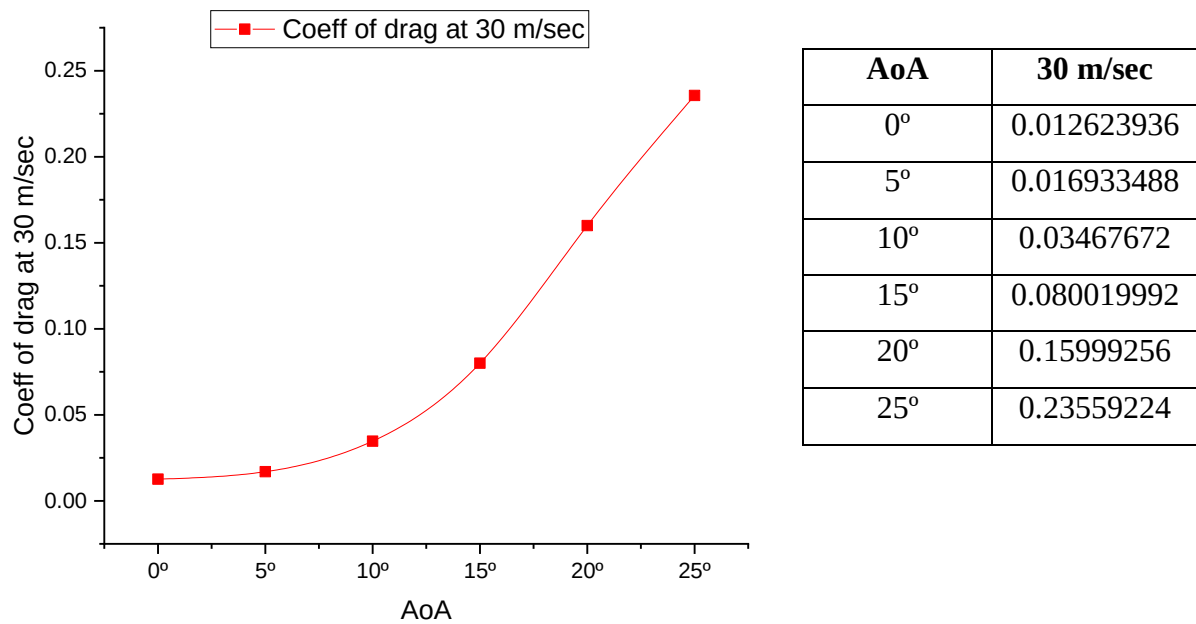


Fig 4-23 Coefficient of drag Vs AoA at 30 m/sec for variable cambered NACA 23012

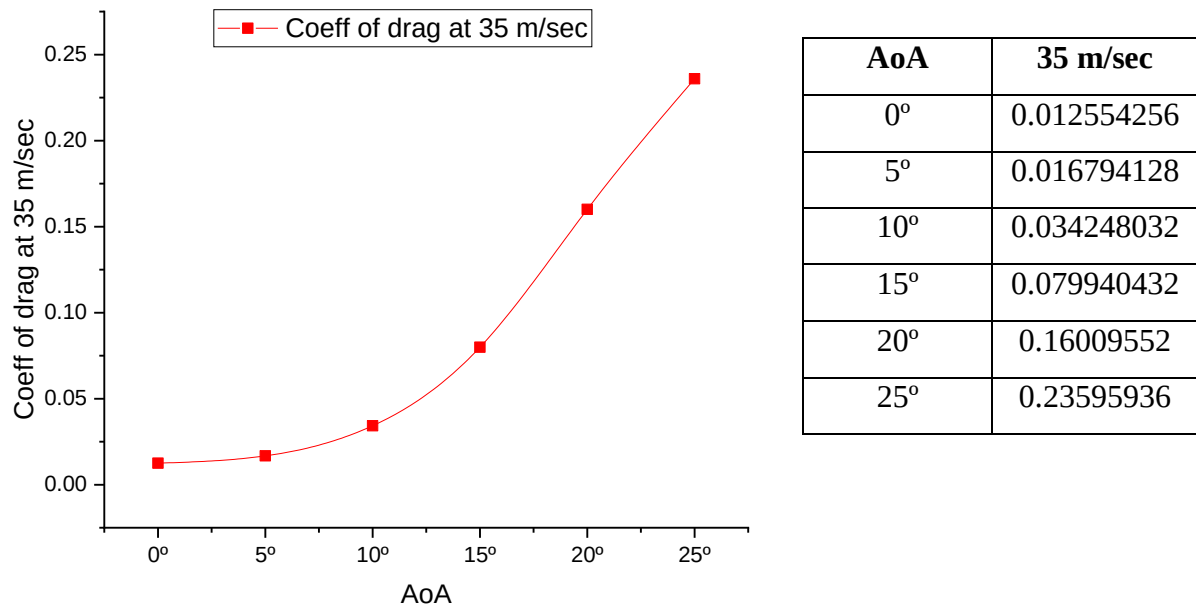


Fig 4-24 Coefficient of drag Vs AoA at 35 m/sec for variable cambered NACA 23012

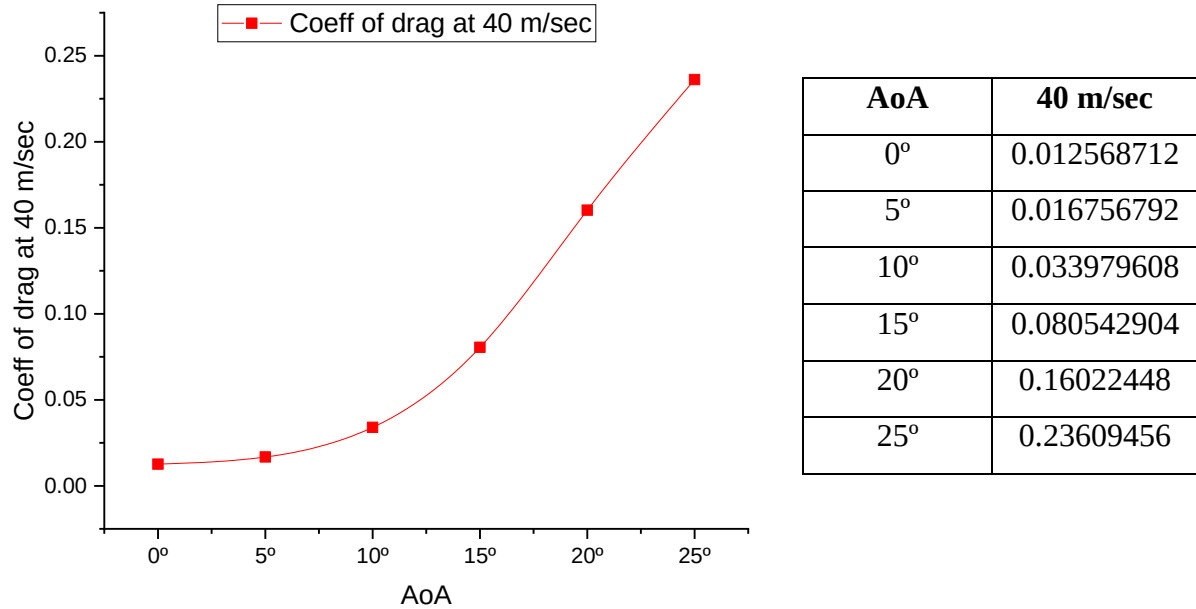


Fig 4-25 Coefficient of drag Vs AoA at 40 m/sec for variable cambered NACA 23012

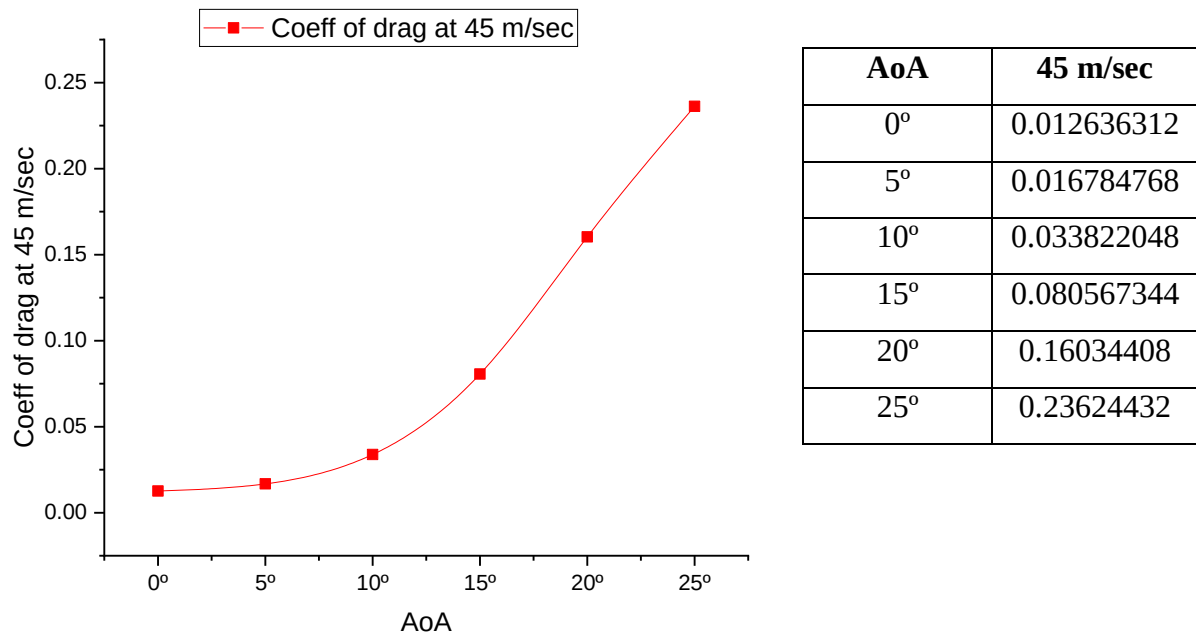


Fig 4-26 Coefficient of drag Vs AoA at 45 m/sec for variable cambered NACA 23012

With the help of a subsonic wind tunnel, the values of coefficient of lift and coefficient of drag at a different angles of attack and various velocities are noted, and the graphs are plotted. And similarly, Ansys design and simulation were conducted on the same NACA 23012 variable camber airfoil and data was collected and noted down, the designed NACA 23012 airfoil series has a span of 45 cm and chord of 19 cm, and with the variable camber mechanism, the chord length increases to 21cm. During the experiment, there was an increase in the coefficient of lift and drag values for variable camber NACA series. Now the next objective is to plot computational data for NACA 23012 and variable camber NACA 23102 and compare the data with existing experimental data and arrive at a conclusion; we calculate the difference in increment in coefficient of lift and coefficient of a drag after comparing both the data will be able to arrive at a better result validation.

# CHAPTER 5

## COMPUTATIONAL DATA ANALYSIS

## Geometry

As the topic itself suggests that the work has been done on the wing model with a NACA series airfoil, NACA 23012. Airfoil coordinates were taken from airfoil tools and were plotted in Ansys Modeller for simulation with a chord length of 1 meter (1 meter is

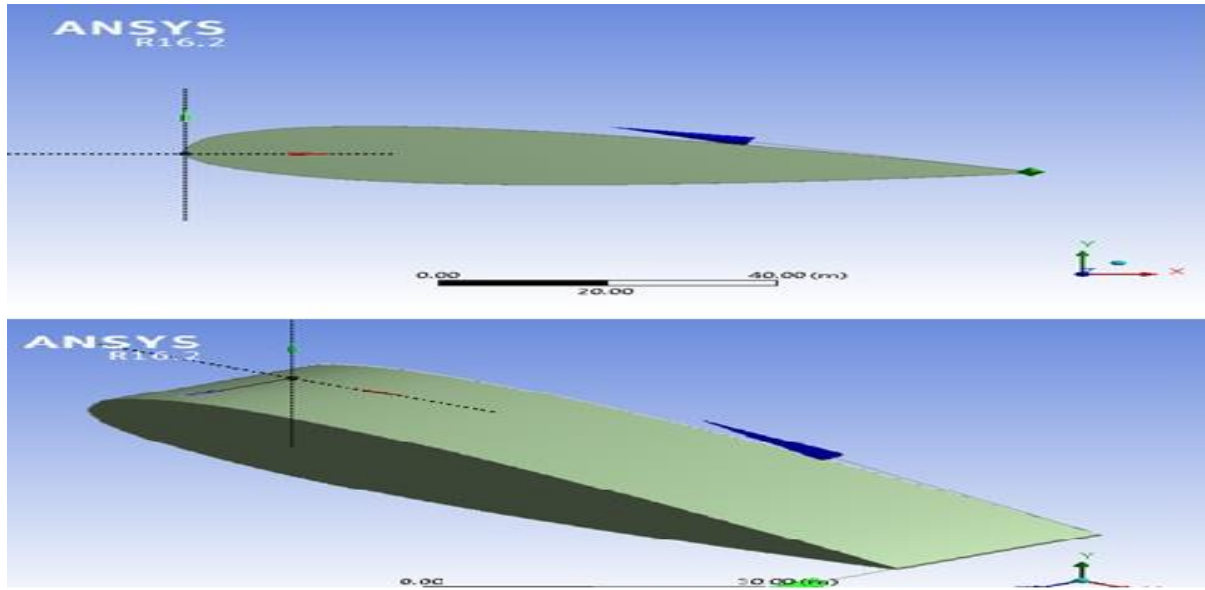


Fig 5-1 Geometry of NACA 23012 series

taken for simplicity in calculations). And a body was created with a simple extrude. The generated airfoil model is as shown in the above fig. Later an outside surrounding (enclosure) was created that will act as a surrounding to the model. The dimensions of the enclosure/surrounding are span 45cm and chord 21 cm. The size of the model is taken with reference to the wing model to obtain the flow dynamics around the model as far as the flow is disturbed. The wing was created as a solid model for which 3D simulation was done. And it was given material of softwood building board for analysis purposes as it accompanies small-scale prototype analysis. For the simulation purpose, the enclosure is made as solid inside which air will flow, and the portion of the wing is kept hollow inside the enclosure. Ultimately the overall aim is to achieve an obstruction inside the flow field on which aerodynamic forces will act.

## Meshing

Meshing is an important prerequisite for any analysis as it determines the quality of the results.

For the same Ansys, mesh was used. Since we wanted the result to be as good as possible, we divided the whole model into some particular sections. And individually defined the meshing parameters for those sets of sections. Model sections where the flow was expected to be complex

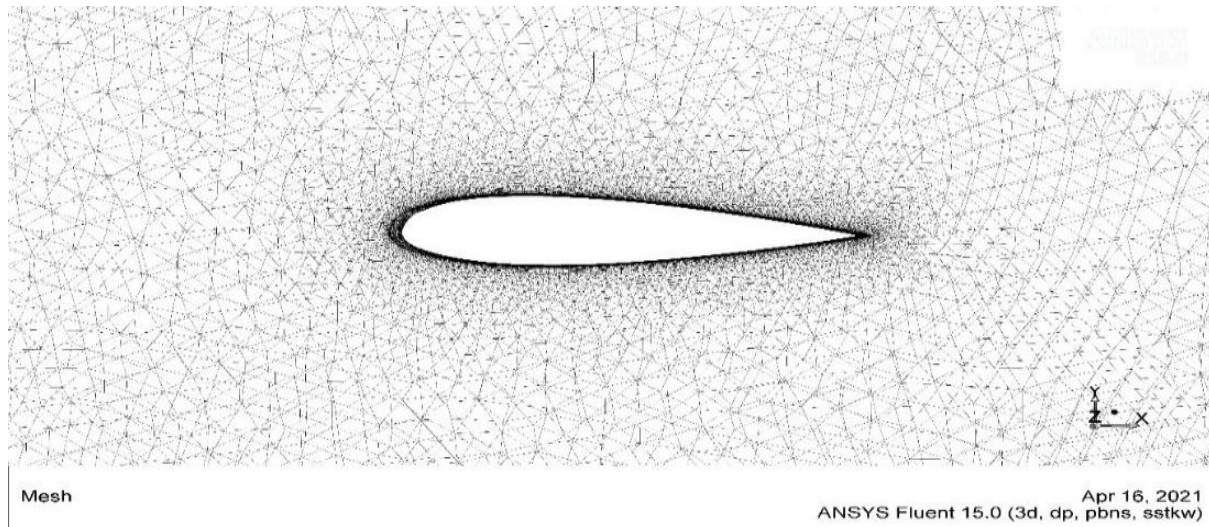


Fig 5-2 Meshing of the aerofoil structure

were refined smoothly, and sections with the less complex flow were refined with coarse specifications. A 2D view of the airfoil is shown in fig. \_\_. We wanted the detailed flow analysis near the boundaries of the airfoil; thus, meshes with small nodes and with higher refinement were created near the boundaries. Also, it would help to analyze the boundary layer much better, as boundary layer formation is an important aspect to watch. And it determines the stall characteristics in an airfoil.

The overall number of elements and nodes generated for the whole model is 15,85,746 and 2,58,951. We tried to have as many sections that our system could support and work on. every section creates a particular control volume where the flow variable is created. And a more the number of sections better is the result. Moreover, we got an average mesh metric orthogonal quality equal to 1, which determines that it can hold as good as the conservation of momentum and conservation of mass relations

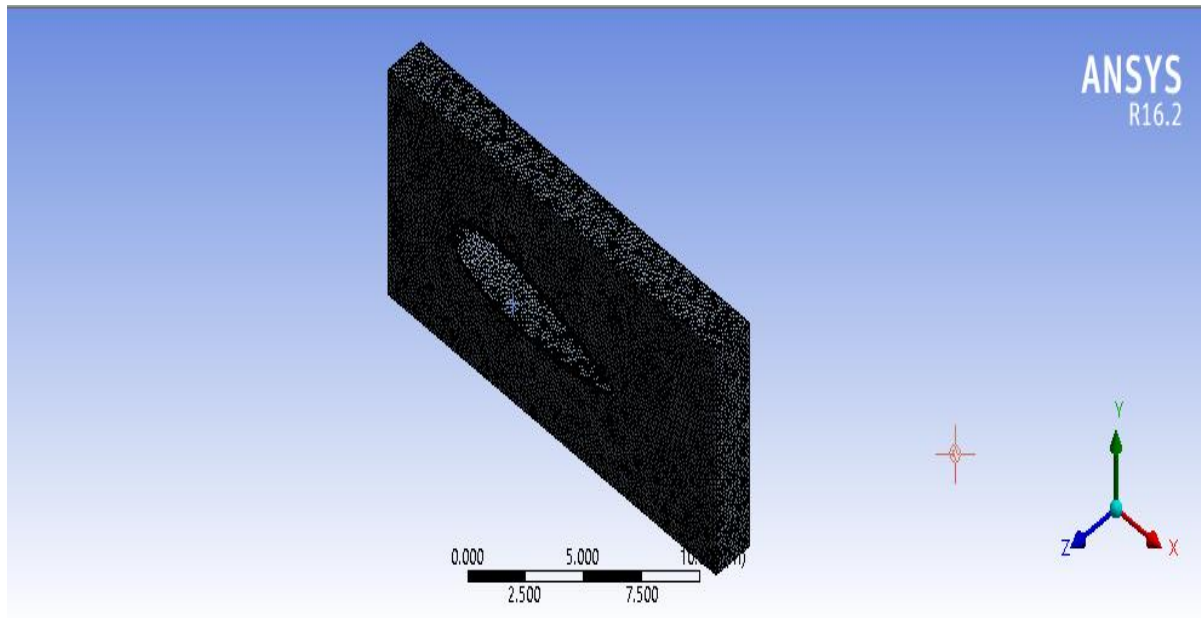


Fig 5-3 illustrates a 3D image of the mesh of the model.

Some other details of the mesh area – Airfoil edge sizing was also defined to be 300 sections on the of the airfoil. And body sizing element size was set as 0.05 meters. About the inflation, it is specified as ten cells of layers. Each at a distance of multiple of 1.2 to every predecessor layer distance from the boundary. The maximum distance of formation of inflation layers was set as 0.01 meters away from the boundary with the second level of refinement.

### **Analysis**

After all the prerequisites were done for Ansys fluent, the mesh geometry was imported into Fluent, where a later setup was done. In Ansys fluent, the boundaries of the simulation were specified as one edge was selected as the inlet of the flow with adiabatic heat transfer settings, and the opposite with output of the flow with relative pressure as zero Pascal's.

The other two sides were specified as slip walls. Slip walls was selected because there will be no boundary layer formation on the walls, and the surrounding can be better compared with real surroundings. And the edges of the airfoil were selected as no-slip walls. Which will act as an object in the airflow on which the test is being performed. The domain of the complete enclosure with the object is shown in fig. \_\_\_\_ with inlets and outlets as a 2D image. And fig\_\_ is shown for the 3D view of the flow parameters and settings. In the flow regime, the air was specified for the flow with a velocity of 20 m/s. Which is a subsonic value at 25 degrees Celsius. The reference

pressure of the flow system is considered as 1 atm sea level air pressure with the object being stationary inside the flow field. Here, the setting is such that inside the flow field, air will move all around the airfoil. The airfoil, being static/stationary, will experience different forces around it, which will be analyzed plotted in various graphs.

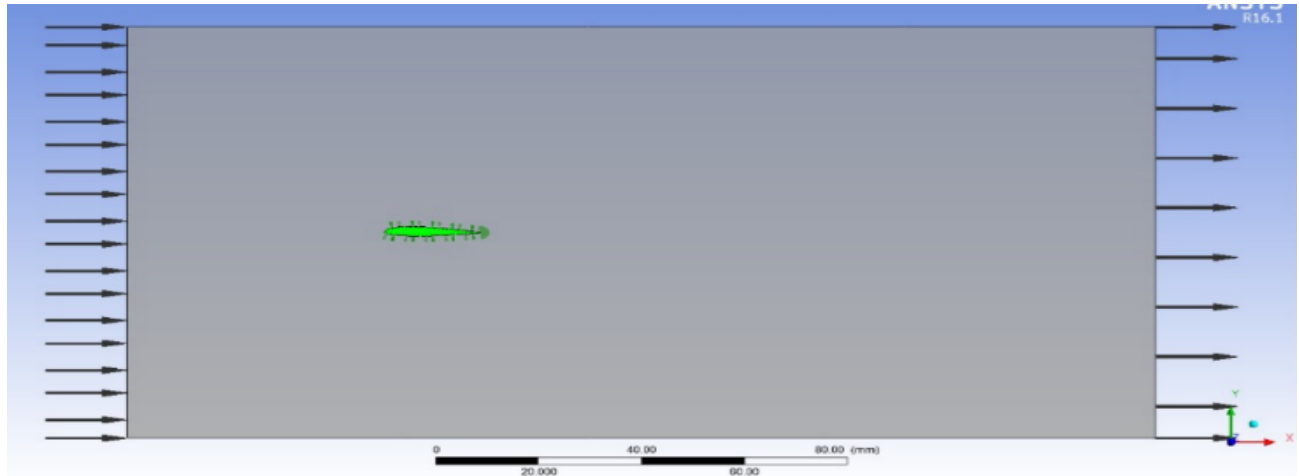


Fig 5-4 Aerofoil placed inside the control volume (slip walls)

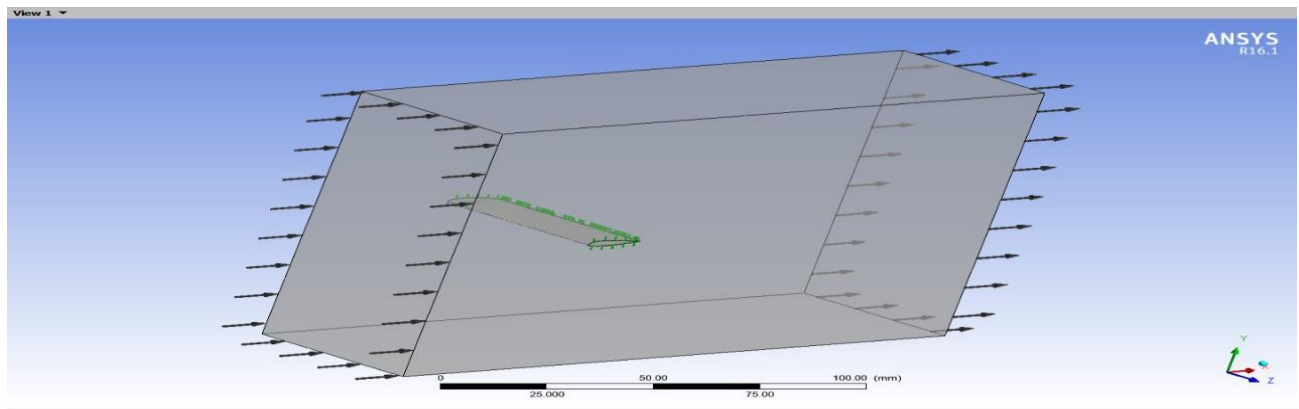


Fig 5-5 Aerofoil inside the slip walls at different AoA

The turbulence model was set as k-Lipson. This is one of the most common turbulence models used in flow simulations in turbulent flow conditions. This model is a two-equation model that provides a general description of turbulence using two transport equations.

## **Analysis**

The fluent Domain is discretized using the Finite volume method, and the boundary condition is as follows

- General – Pressure Based Model for better results.
- Models–Viscous Laminar-K-Epsilon relation for solving the finite volume.
- Materials – Internal material as air
- Cell zone condition – over the surface body
- Boundary Conditions – Velocity inlet and pressure Outlet are the main boundary condition with in which the different inlet velocities of 20 m/sec,25 m/sec,30 m/sec,35 m/sec,40 m/sec,45 m/sec are considered
- Reference Values – Give the values of pressure, temperature and density at that altitude from ISA.
- Solution Method– Coupled method is followed and solved using the Gauss elimination method.
- Solution Monitors – Cl and Cd Plots have been monitored.
- Solution Initialization – Standard initialization was used
- Solution: - go for the solution and check the plots

### **5.1.6 Ansys Simulation data for NACA 23012**

The following graphs are plotted with the values received for respective speed, and angle of attack. The graphs are plotted for NACA 23012 and variable camber NACA 23012 airfoil; here in this report, coefficient of lift as well as coefficient of drag is plotted separately.

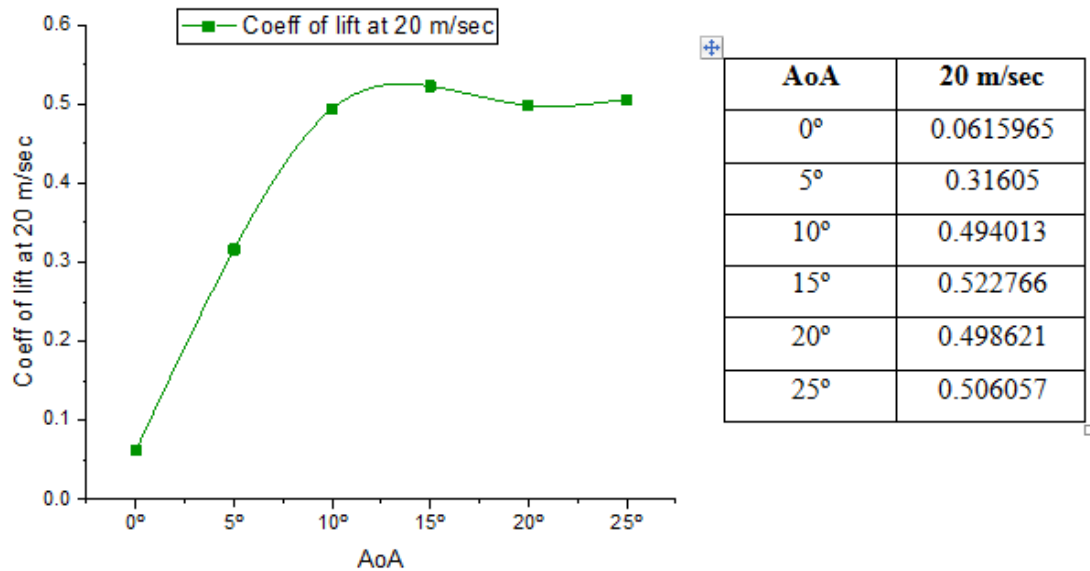


Fig 5-6 Coefficient of lift Vs AoA at 20 m/sec for NACA 23012

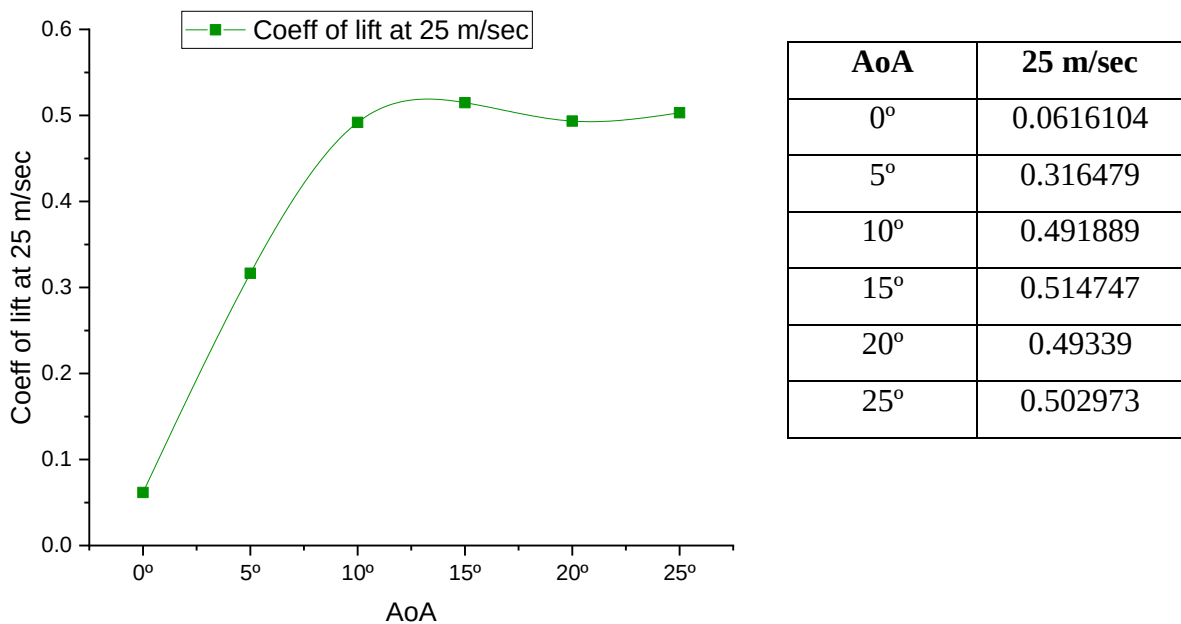
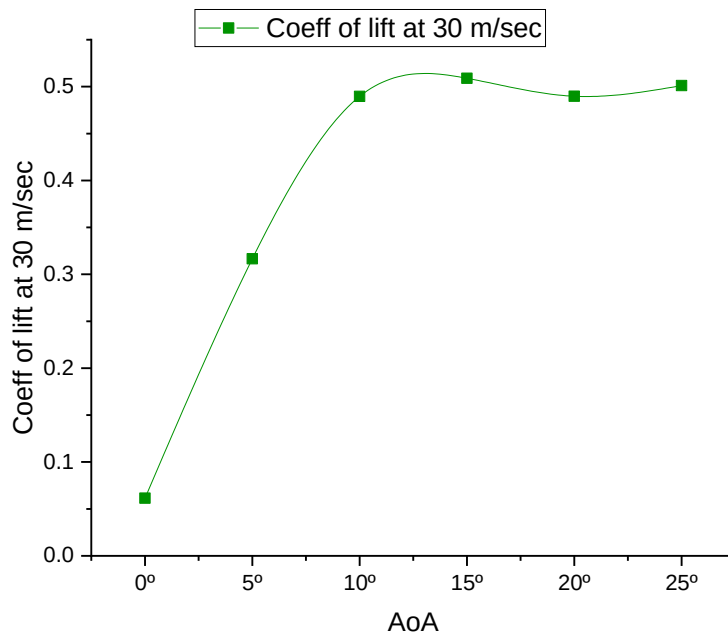
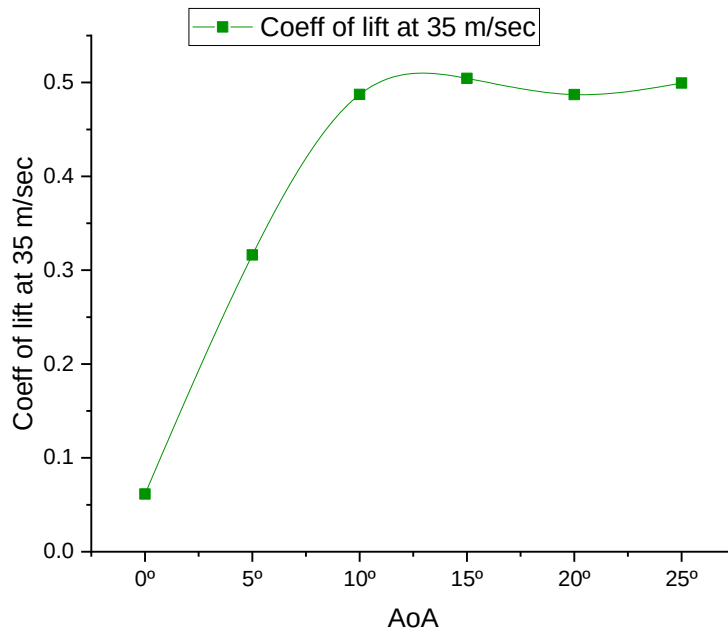


Fig 5-7 Coefficient of lift Vs AoA at 25 m/sec for NACA 23012



| AoA | 30 m/sec |
|-----|----------|
| 0°  | 0.061535 |
| 5°  | 0.316488 |
| 10° | 0.489612 |
| 15° | 0.50889  |
| 20° | 0.489762 |
| 25° | 0.501149 |

Fig 5-8 Coefficient of lift Vs AoA at 30 m/sec for NACA 23012



| AoA | 35 m/sec  |
|-----|-----------|
| 0°  | 0.0614103 |
| 5°  | 0.316273  |
| 10° | 0.487297  |
| 15° | 0.504354  |
| 20° | 0.487058  |
| 25° | 0.499408  |

Fig 5-9 Coefficient of lift Vs AoA at 35 m/sec for NACA 23012

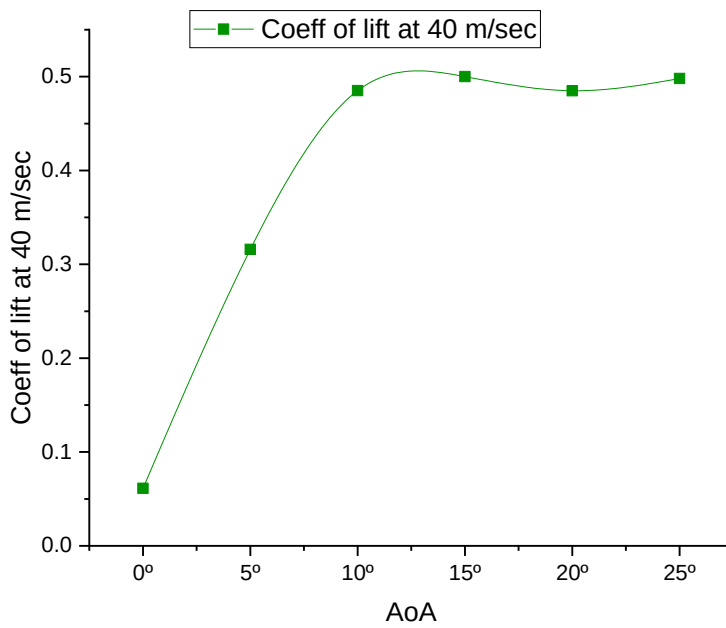


Fig 5-10 Coefficient of lift Vs AoA at 40 m/sec for NACA 23012

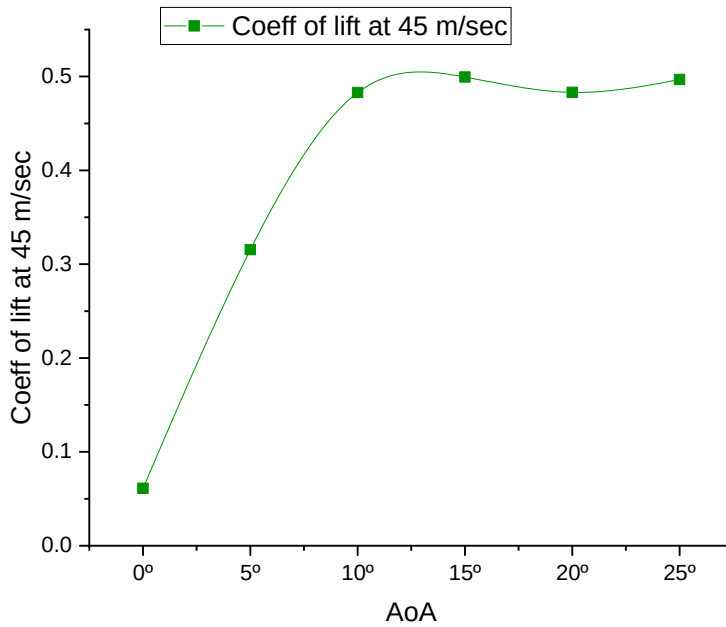


Fig 5-11 Coefficient of lift Vs AoA at 45 m/sec for NACA 23012

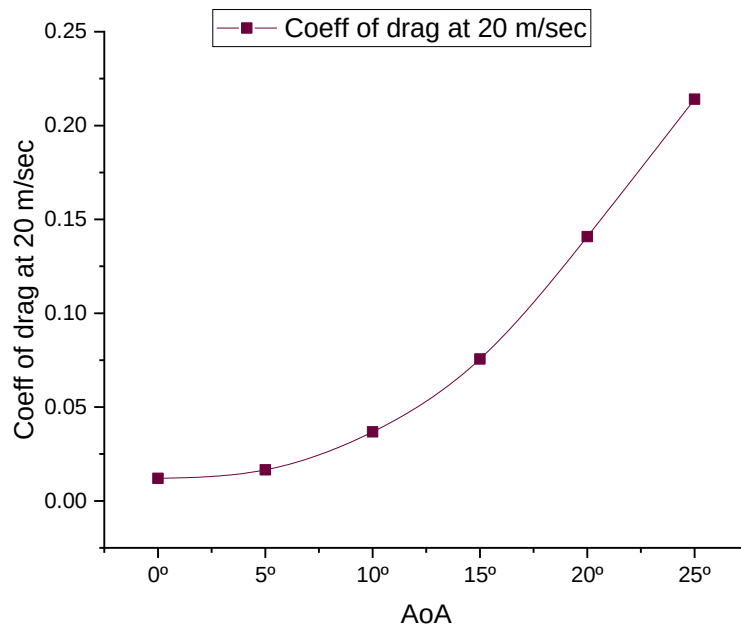


Fig 5-12 Coefficient of drag Vs AoA at 20 m/sec for NACA 23012

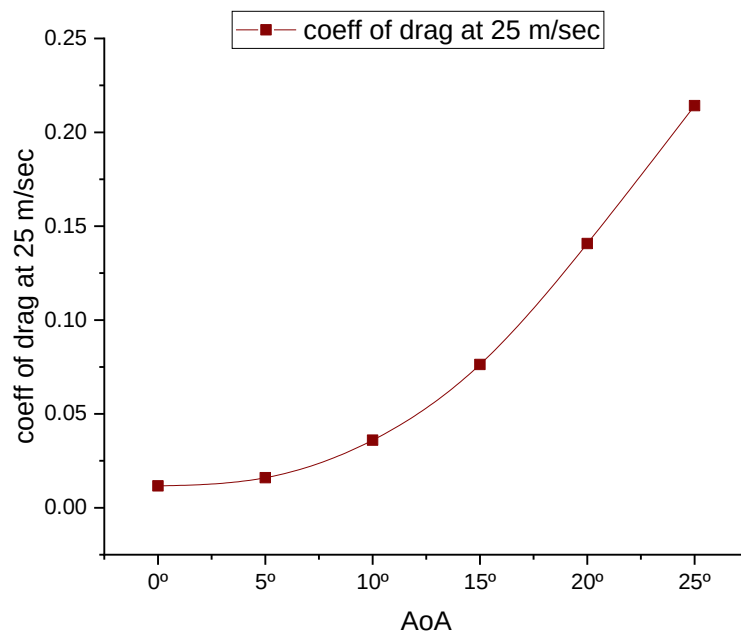


Fig 5-13 Coefficient of drag Vs AoA at 25 m/sec for NACA 23012

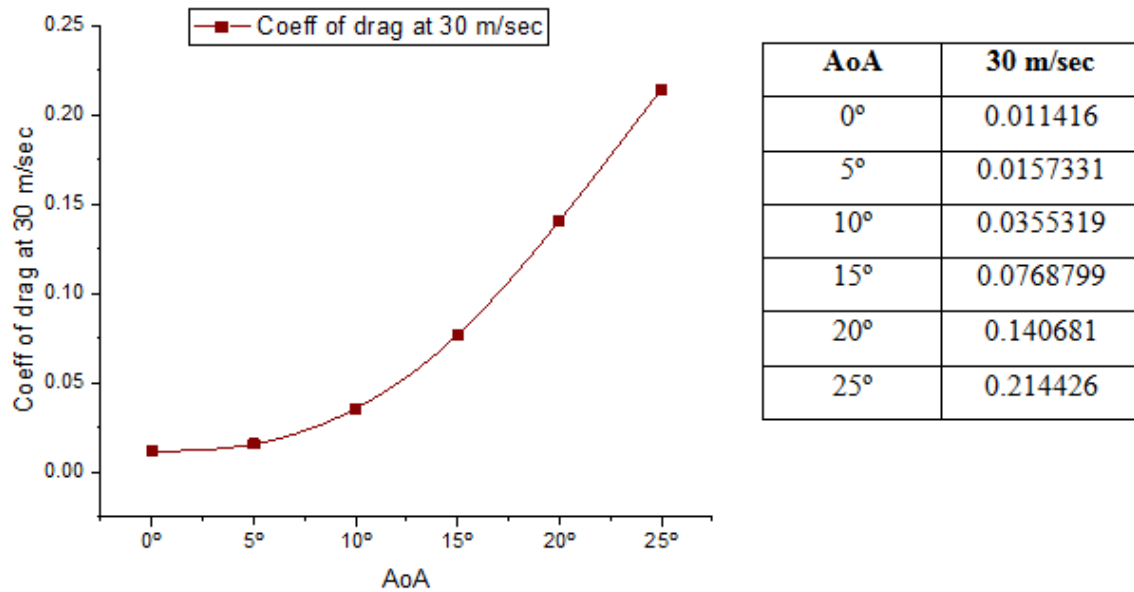


Fig 5-14 Coefficient of drag Vs AoA at 30 m/sec for NACA 23012

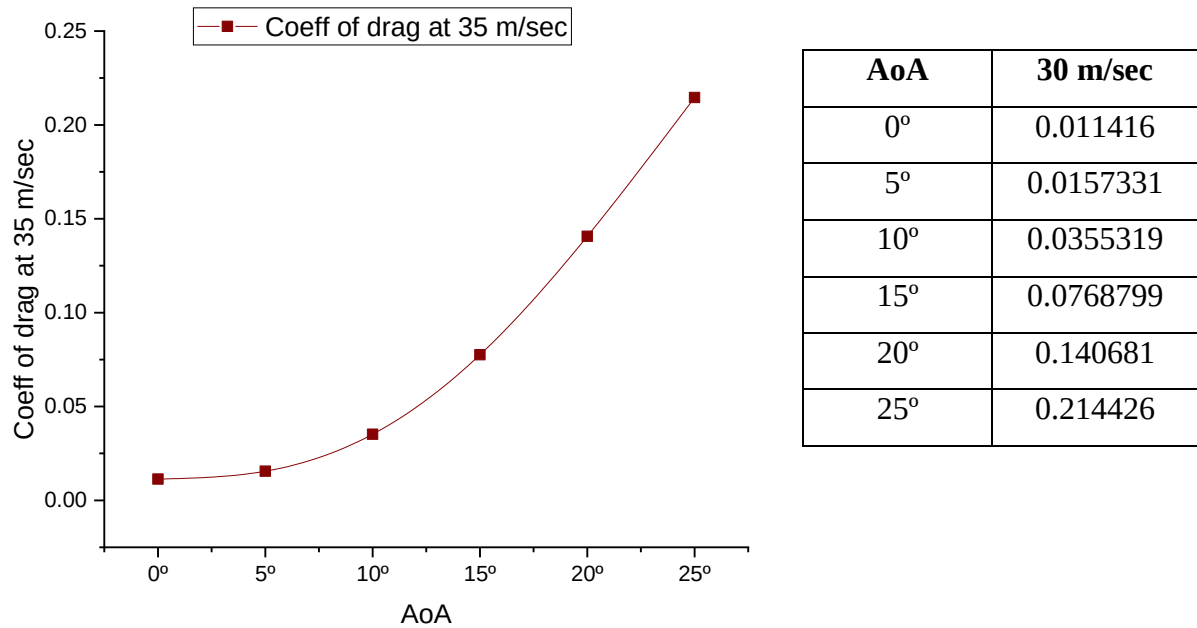


Fig 5-15 Coefficient of drag Vs AoA at 35 m/sec for NACA 23012

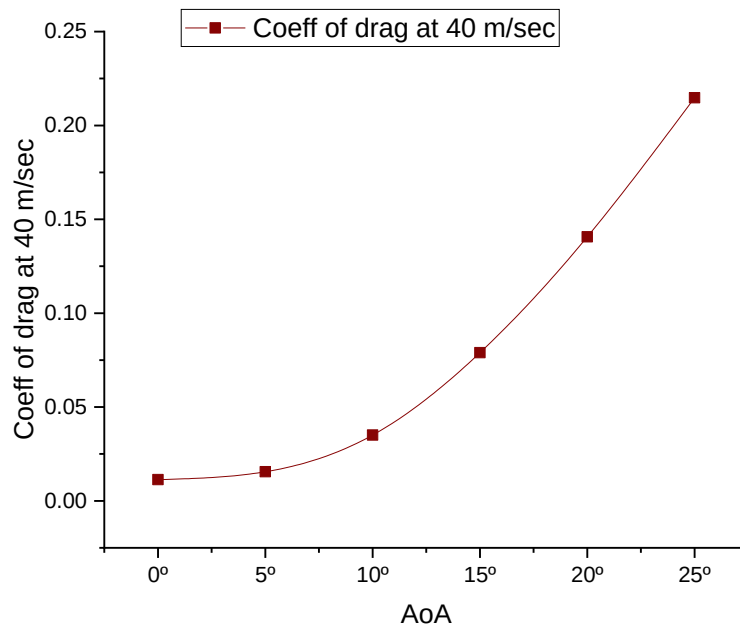


Fig 5-16 Coefficient of drag Vs AoA at 40 m/sec for NACA 23012

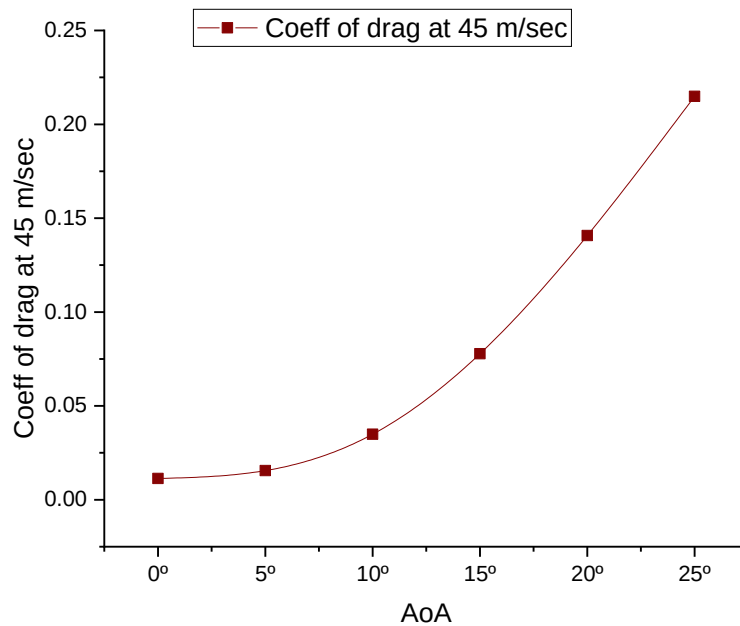


Fig 5-17 Coefficient of drag Vs AoA at 45 m/sec for NACA 23012

### 5.1.7 Ansys simulation data for variable camber NACA 23012

Here the data is plotted for variable camber NACA 23012, keeping the angle of attack, and speed same as we have done for the above NACA Series.

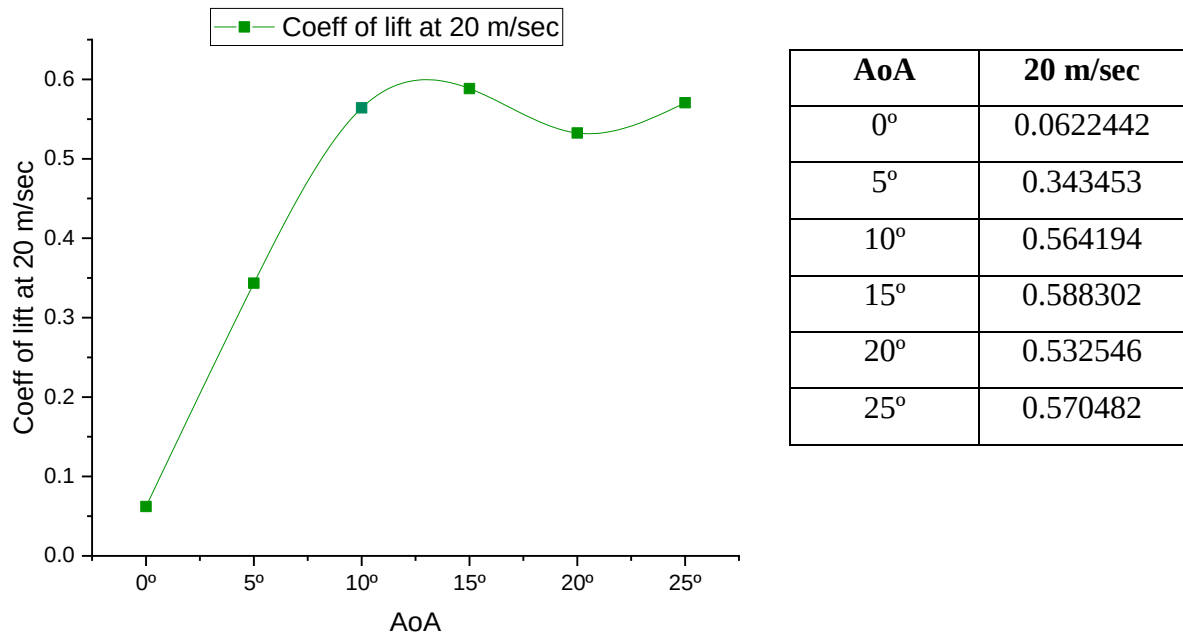


Fig 5-18 Coefficient of lift Vs AoA at 20 m/sec for variable camber NACA 23012

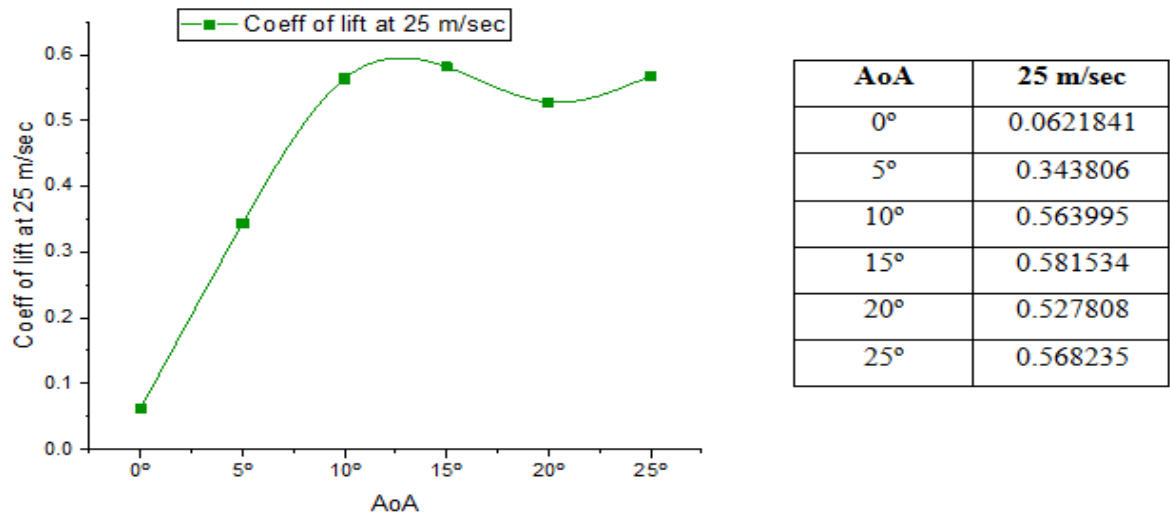


Fig 5-19 Coefficient of lift Vs AoA at 25 m/sec for variable camber NACA 23012

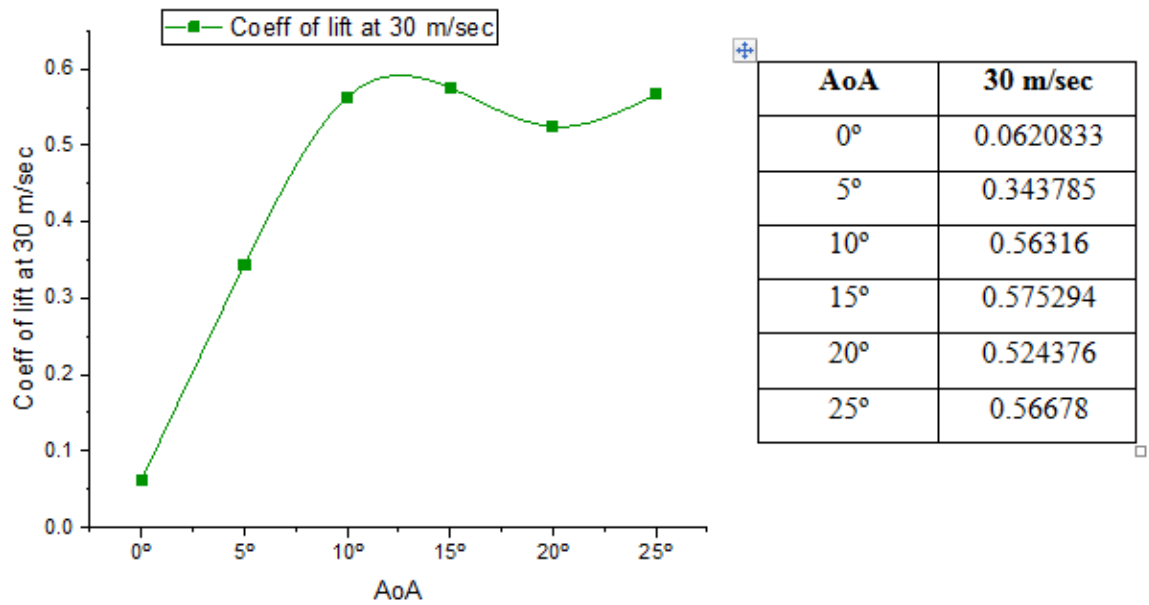


Fig 5-20 Coefficient of lift Vs AoA at 30 m/sec for variable camber NACA 23012

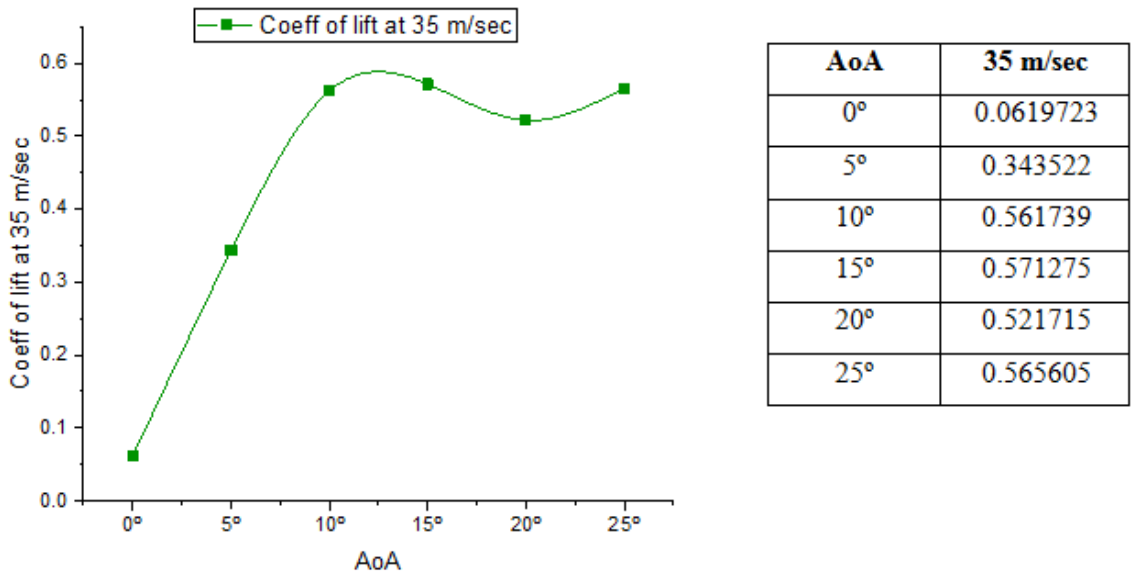
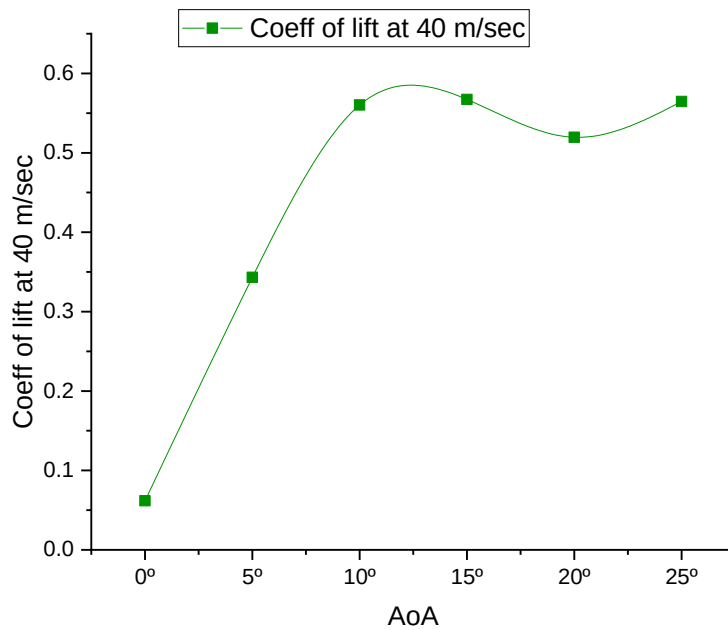
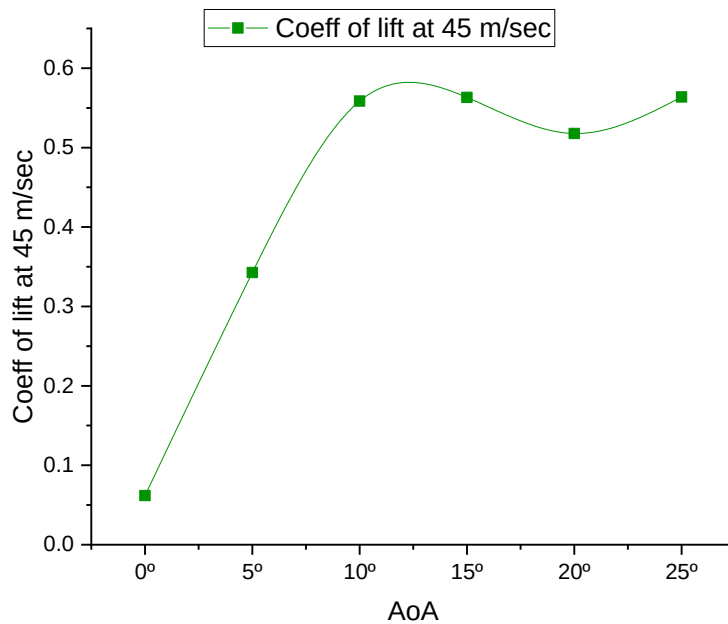


Fig 5-21 Coefficient of lift Vs AoA at 35m/sec for variable camber NACA 23012



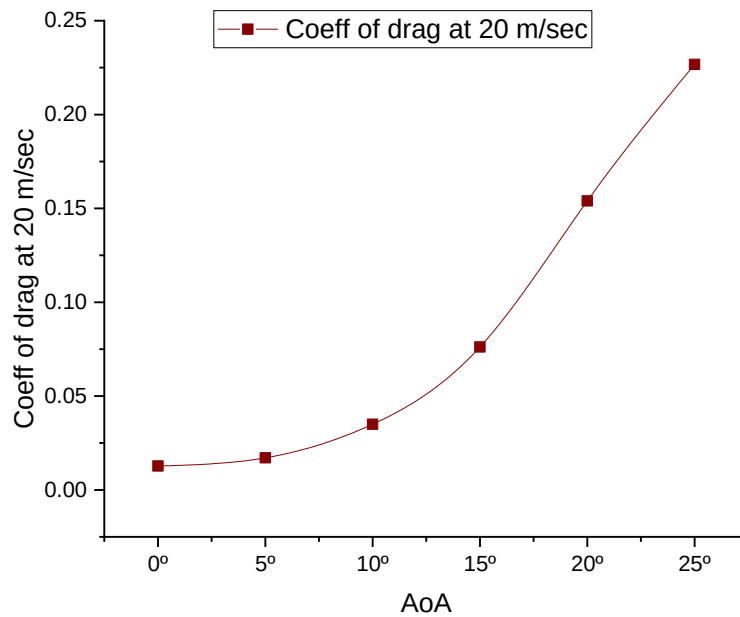
| AoA | 40 m/sec |
|-----|----------|
| 0°  | 0.061863 |
| 5°  | 0.343122 |
| 10° | 0.560194 |
| 15° | 0.567212 |
| 20° | 0.519539 |
| 25° | 0.564665 |

Fig 5-22 Coefficient of lift Vs AoA at 40 m/sec for variable camber 23012



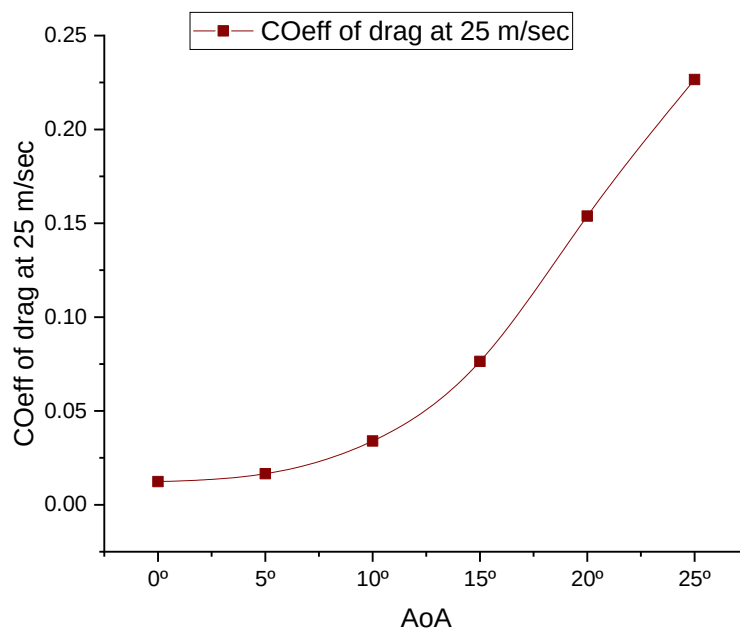
| AoA | 45 m/sec  |
|-----|-----------|
| 0°  | 0.0617578 |
| 5°  | 0.342688  |
| 10° | 0.558582  |
| 15° | 0.563265  |
| 20° | 0.517688  |
| 25° | 0.563824  |

Fig 5-23 Coefficient of lift Vs AoA at 45 m/sec for variable camber 23012



| AoA | 20 m/sec  |
|-----|-----------|
| 0°  | 0.0127174 |
| 5°  | 0.017102  |
| 10° | 0.0350149 |
| 15° | 0.0761779 |
| 20° | 0.153957  |
| 25° | 0.226695  |

Fig 5-24 Coefficient of drag Vs AoA at 20 m/sec for variable camber NACA 23012



| AoA | 25 m/sec  |
|-----|-----------|
| 0°  | 0.0123297 |
| 5°  | 0.0165693 |
| 10° | 0.0339942 |
| 15° | 0.0764314 |
| 20° | 0.153789  |
| 25° | 0.226581  |

Fig 5-25 Coefficient of drag Vs AoA at 25 m/sec for variable camber NACA 23012

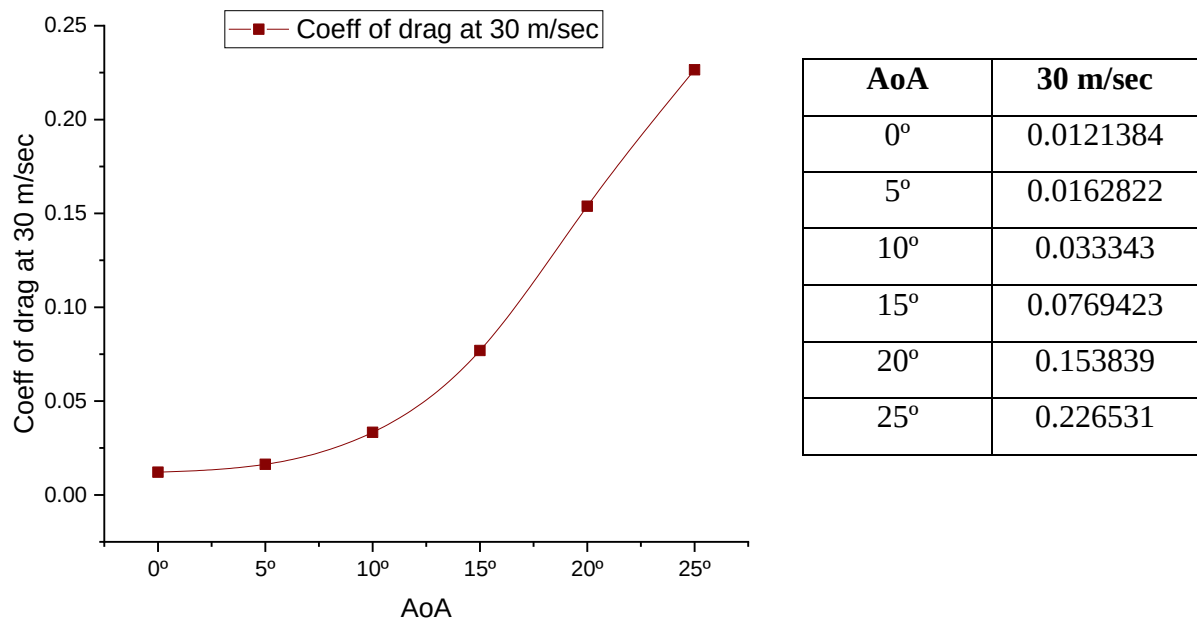


Fig 5-26 Coefficient of drag Vs AoA at 30 m/sec for Variable camber NACA 23012

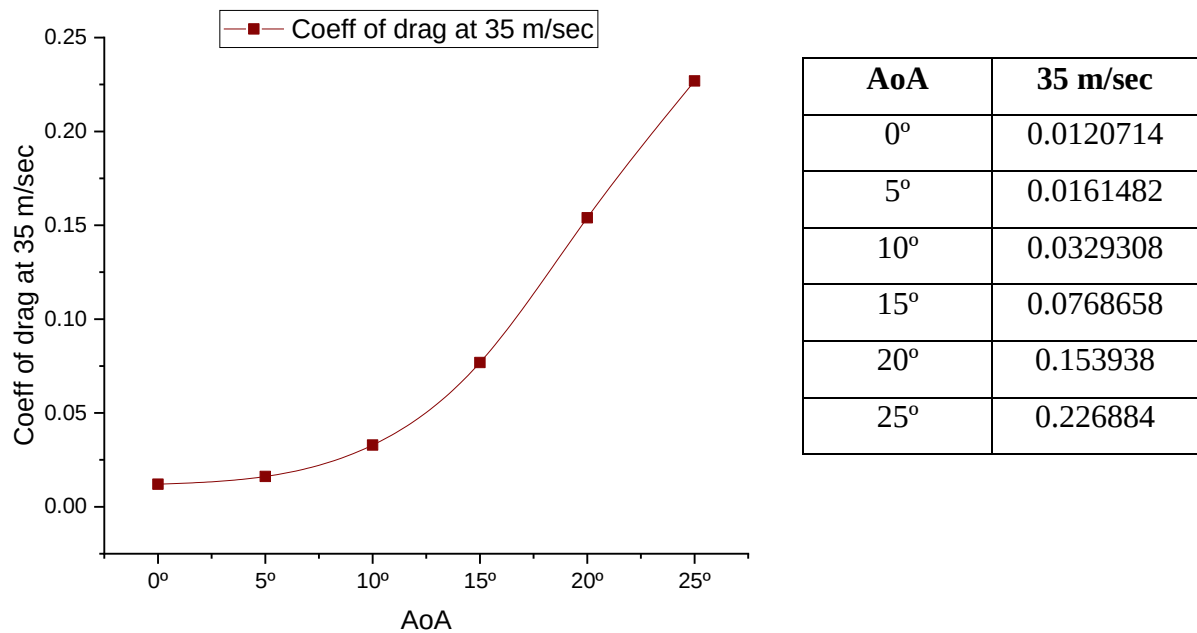


Fig 5-27 Coefficient of drag Vs AoA at 35 m/sec for variable camber NACA 23012

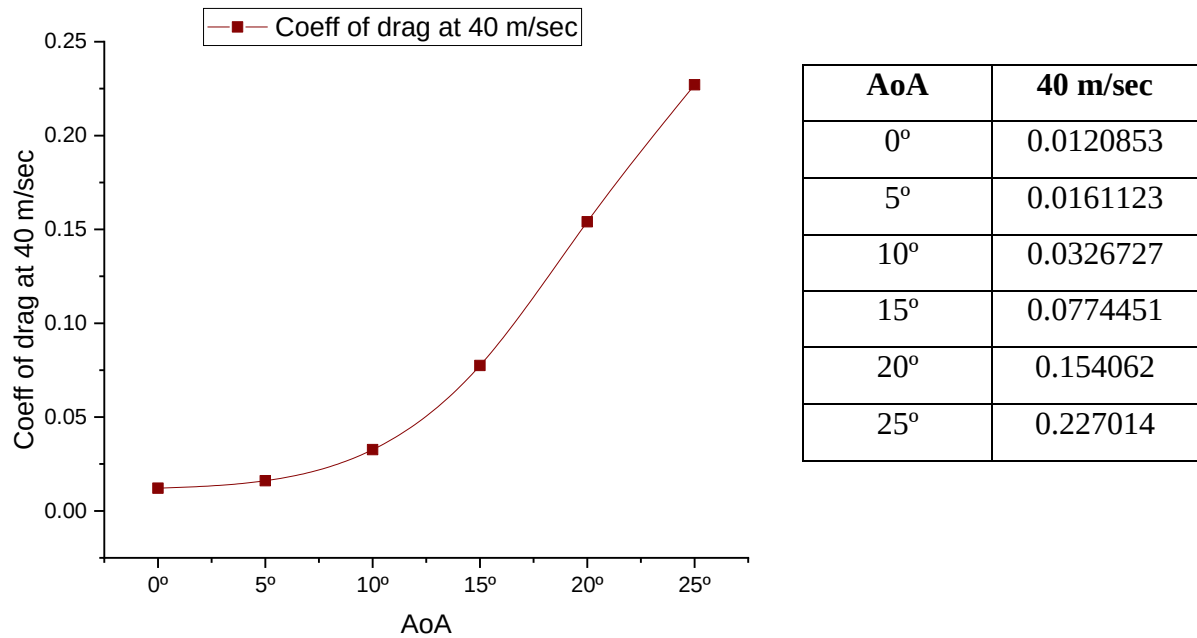


Fig 5-28 Coefficient of drag Vs AoA at 40 m/sec for variable camber NACA 23012

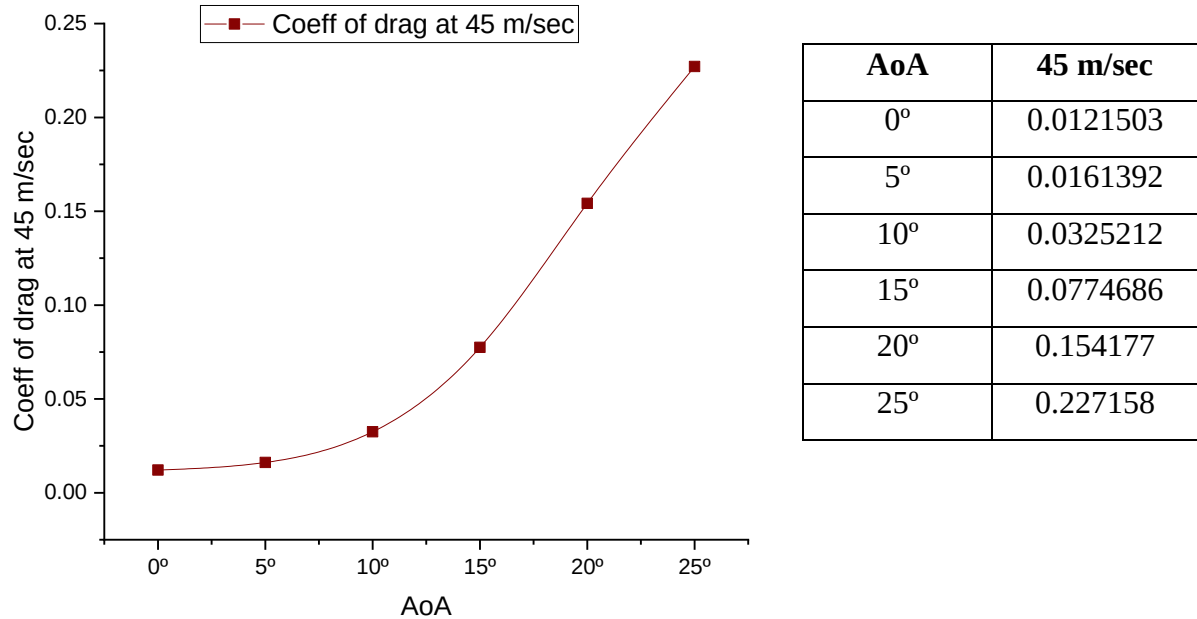


Fig 5-29 Coefficient of drag Vs AoA at 45 m/sec for variable camber NACA 23012

### **Pressure and velocity contour plots with respect to Kutta condition in steady flow**

Kutta condition is a principle applicable to steady flow fluid dynamics, especially when we study aerodynamics which involves bodies with sharp edges, such as the trailing edge of airfoils. It's been named after German mathematician and aerodynamicist Martin Kutta.

According to the Kutta condition, a body with a sharp trailing edge that is moving through a fluid will create a circulation of sufficient strength to hold the rear stagnation point at the trailing edge. Fluid flows around a body with a sharp corner, such as a trailing edge; the Kutta condition refers to the flow pattern in which fluid approaches the corner from above and below, meets at the trailing edge of an airfoil, as per the theory, none of the fluid flows around the trailing edge. The Kutta condition is significant when using the Kutta-Joukowski theorem to calculate the lift created by an airfoil with a sharp trailing edge. The value of circulation of the flow around the airfoil must be that value that would cause the Kutta condition to exist.

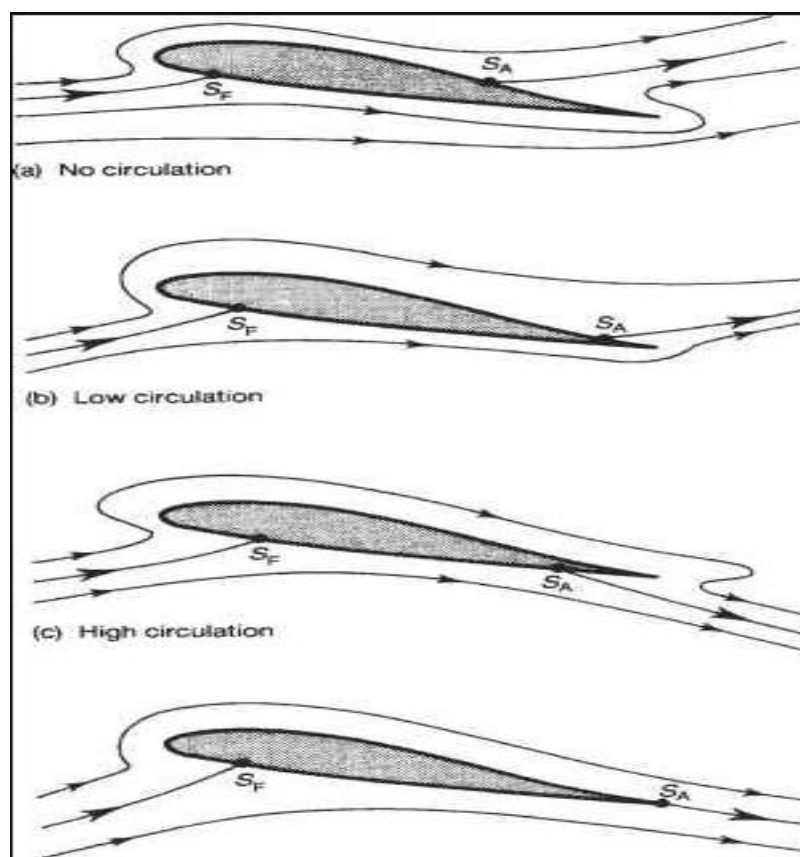


Fig 5-30 Kutta condition for circulation around an airfoil

Suppose an airfoil with a sharp trailing edge begins to move with an angle of attack through the air. In that case, the two stagnation points are initially located on the underside near the leading edge and on the topside near the trailing edge, just as with the cylinder. As the air passing below the airfoil reaches the trailing edge, it must flow around the trailing edge and along the top side of the airfoil toward the stagnation point on the top side of the airfoil. Vortex flow occurs at the trailing edge, and because the radius of the sharp trailing edge is zero, the speed of the air around the trailing edge should be infinitely fast. Though real fluids cannot move at infinite speed, they can move very fast. The high airspeed around the trailing edge causes strong viscous forces to act on the air adjacent to the trailing edge of the airfoil, and the result is that a strong vortex accumulates on the topside of the airfoil, near the trailing edge. As the airfoil begins to move, it carries this vortex, known as the starting vortex. Now with the advancement in understanding and viewing flow regimes and patterns with the help of visualization techniques as well as pressure contour and velocity contours which are plotted while simulating flow around an airfoil in Ansys or any other CFD software, we could exactly see the application of Kutta condition as well as the movement  $S_f$  and  $S_a$  points along with the airfoil profile during the change in velocity and pressure parameter at the various angle of attack, here in this report the change in flow regimes is shown with respect to change in velocity and pressure contour for NACA 23012 as well as variable camber NACA 23012. The pressure contour and velocity contour is plotted for various velocity like 20,25,30,35,40,45 m/sec at a different angle of attack, and the same has been presented below.

## Pressure contour for NACA 23012

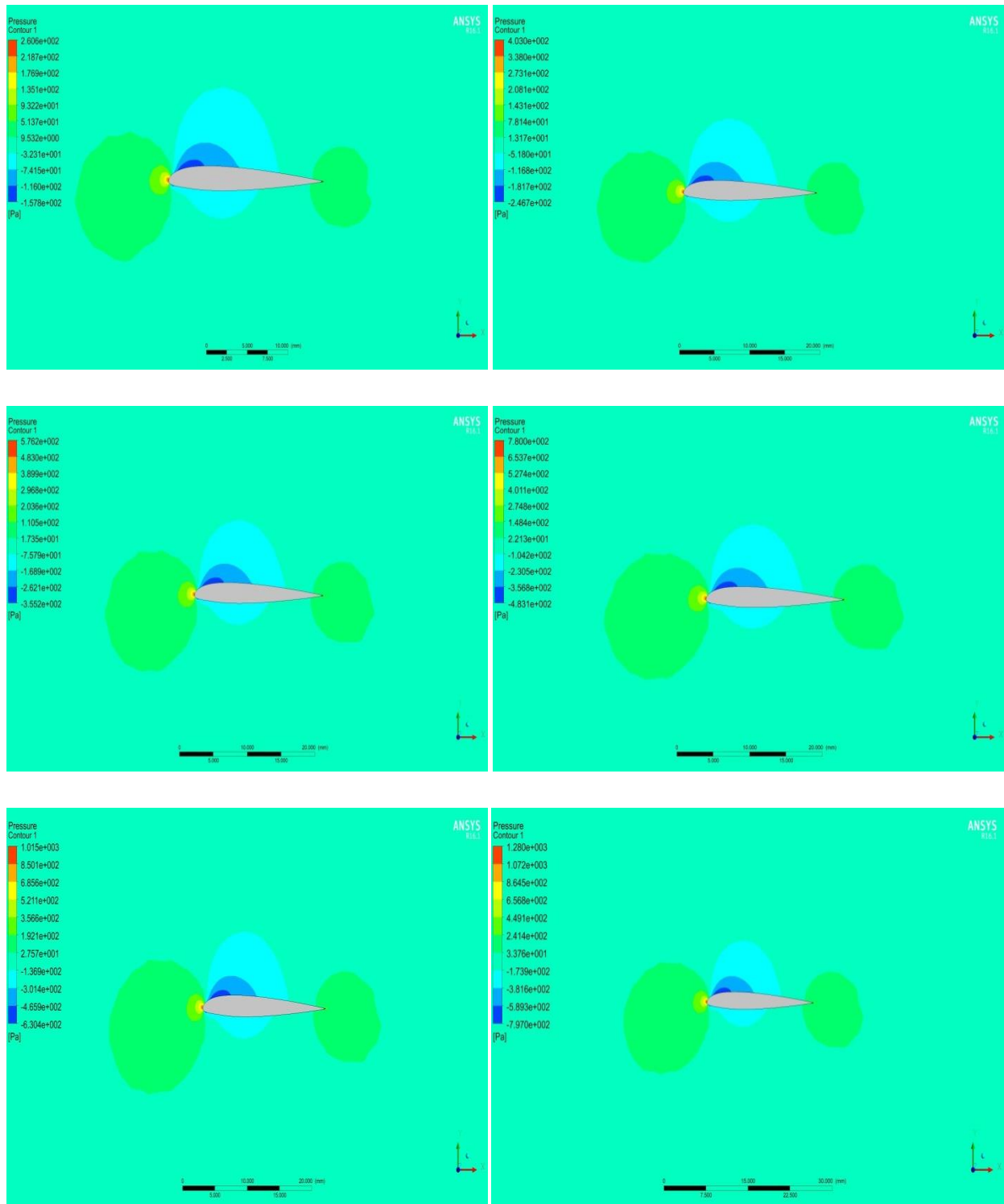


Fig 5-31 Pressure contour plot at various velocity at 20,25,30,35,40,45 m/sec and 0 °AoA for NACA 23012

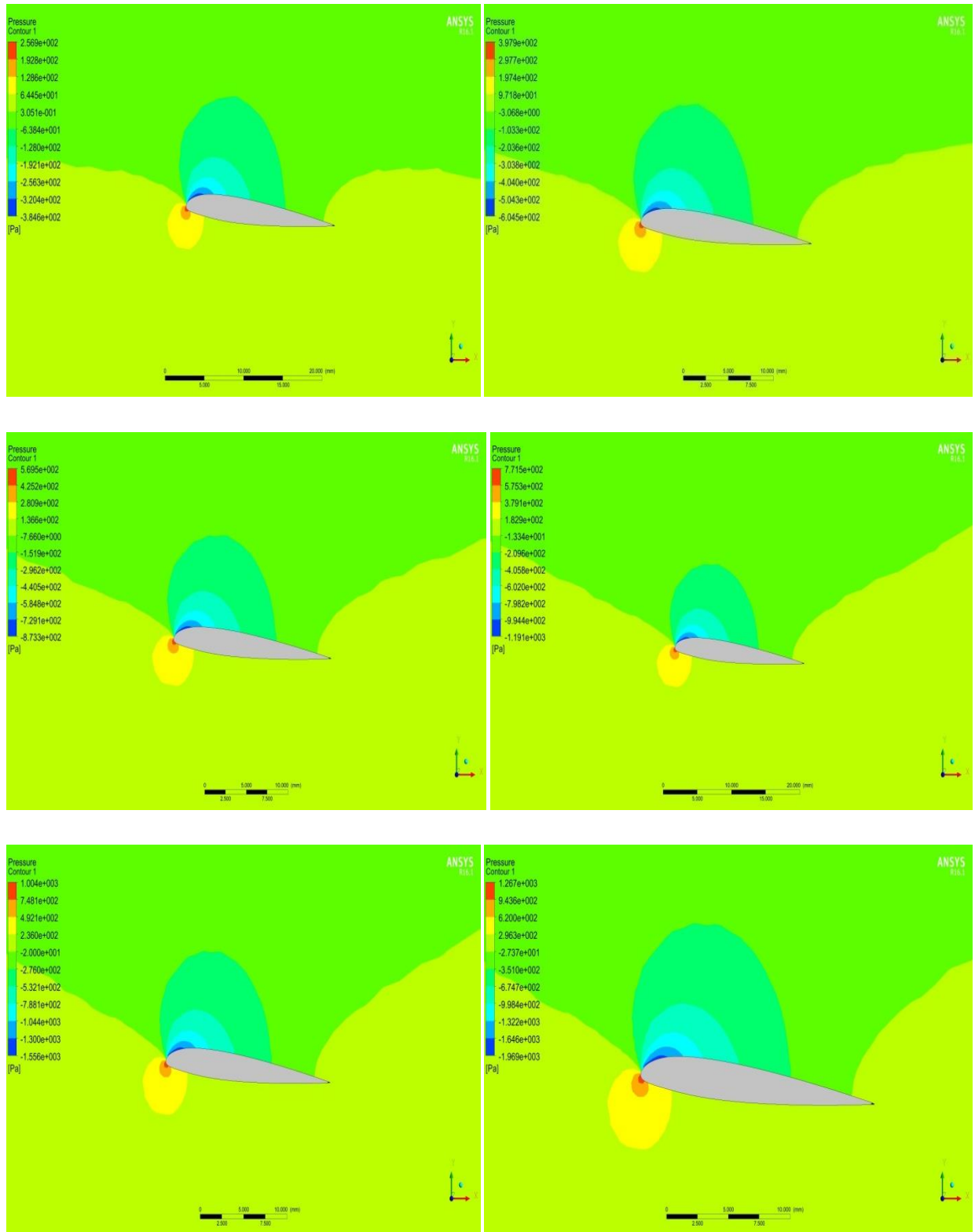


Fig 5-32 Pressure Contour at various velocity at 20,25,30,35,40,45 m/sec and 5° AoA for NACA 23012

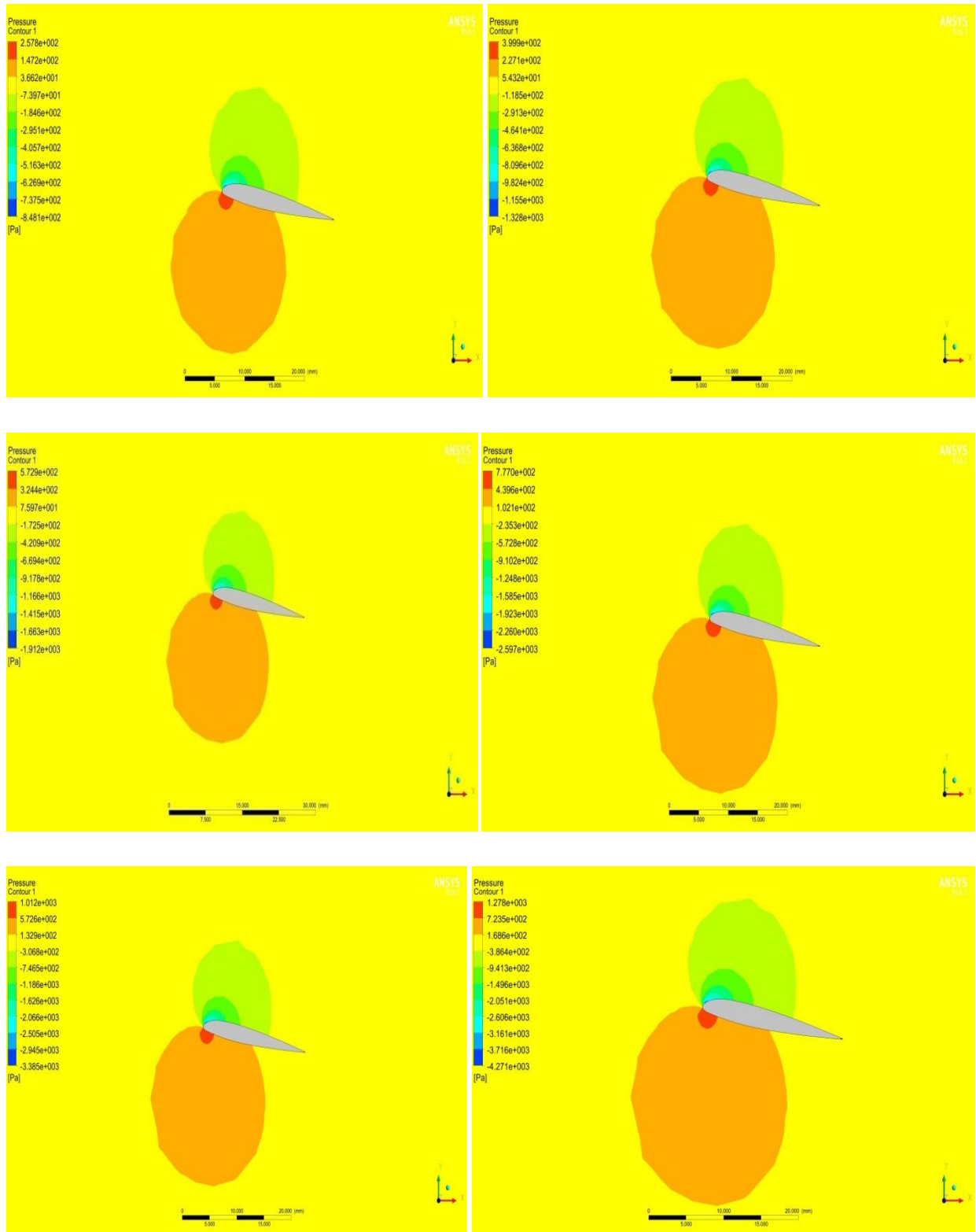


Fig 5-33 Pressure contour at various velocity at 20,25,30,35,40,45 m/sec and 10° AoA for NACA 23012

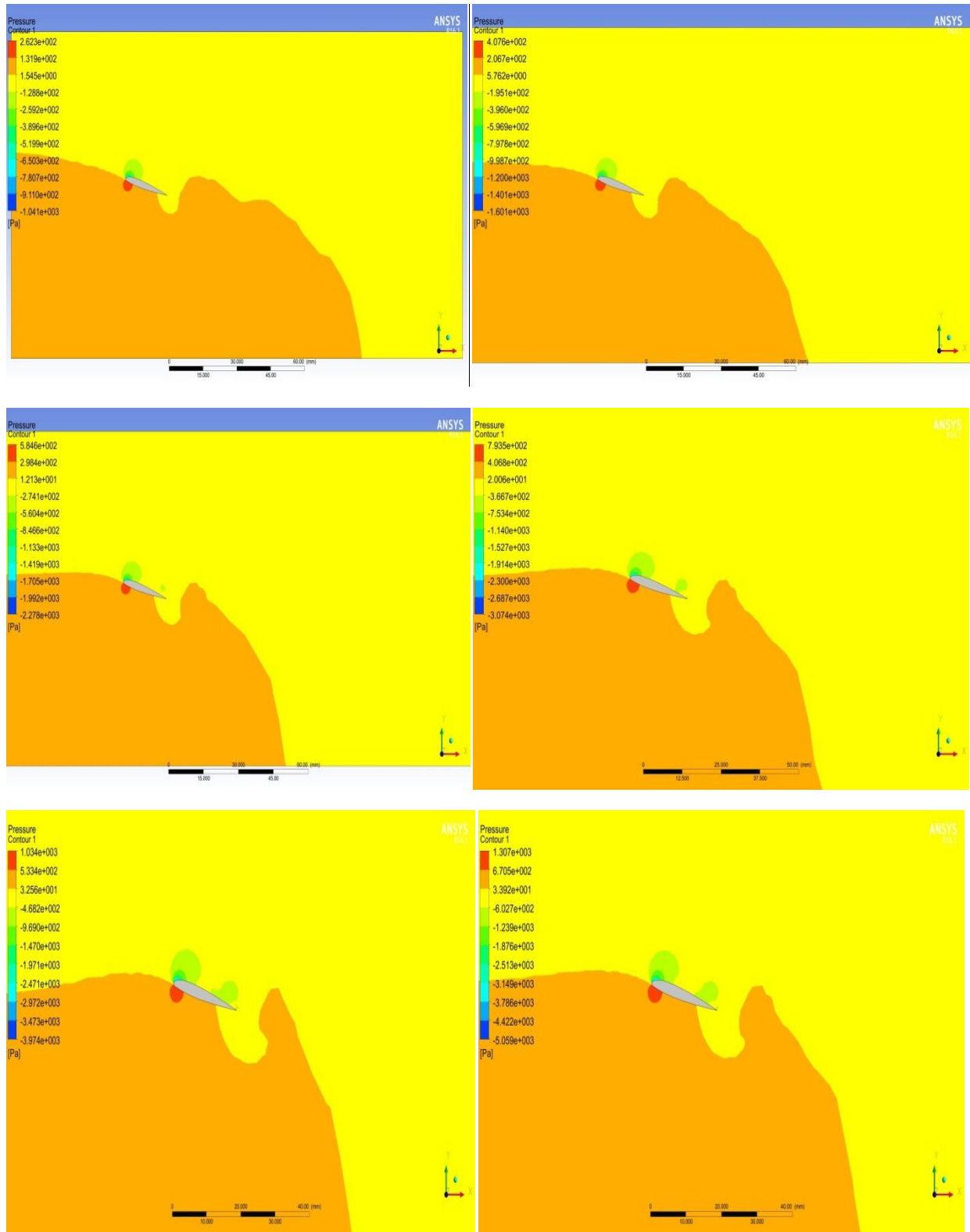


Fig 5-34 Pressure contour at various velocity at 20,25,30,35,40,45 m/sec and 15° AoA for NACA 23012

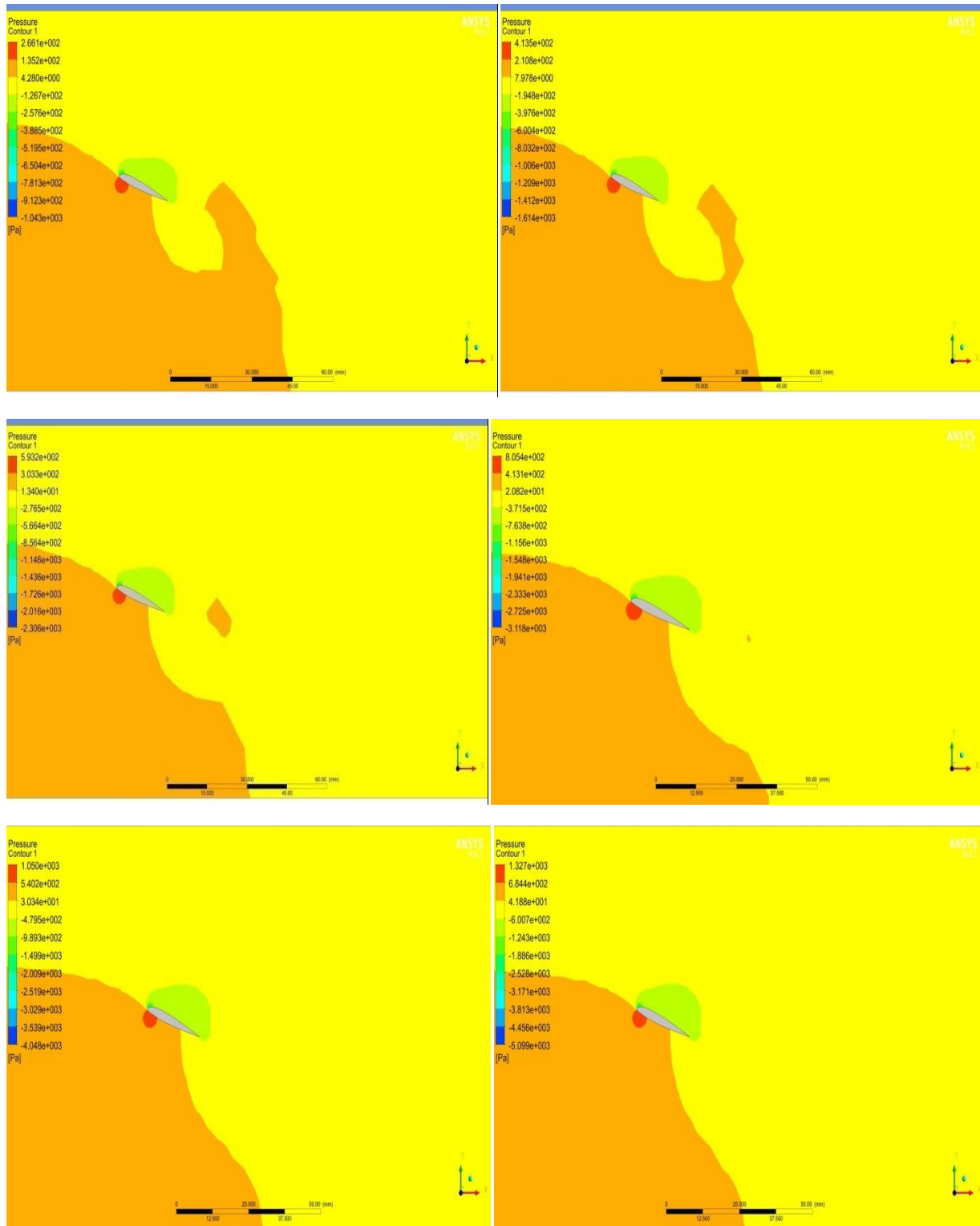


Fig 5-35 Pressure contour plot at various velocities at 20,25,30,35,40,45 m/sec 20° AoA for NACA 23012

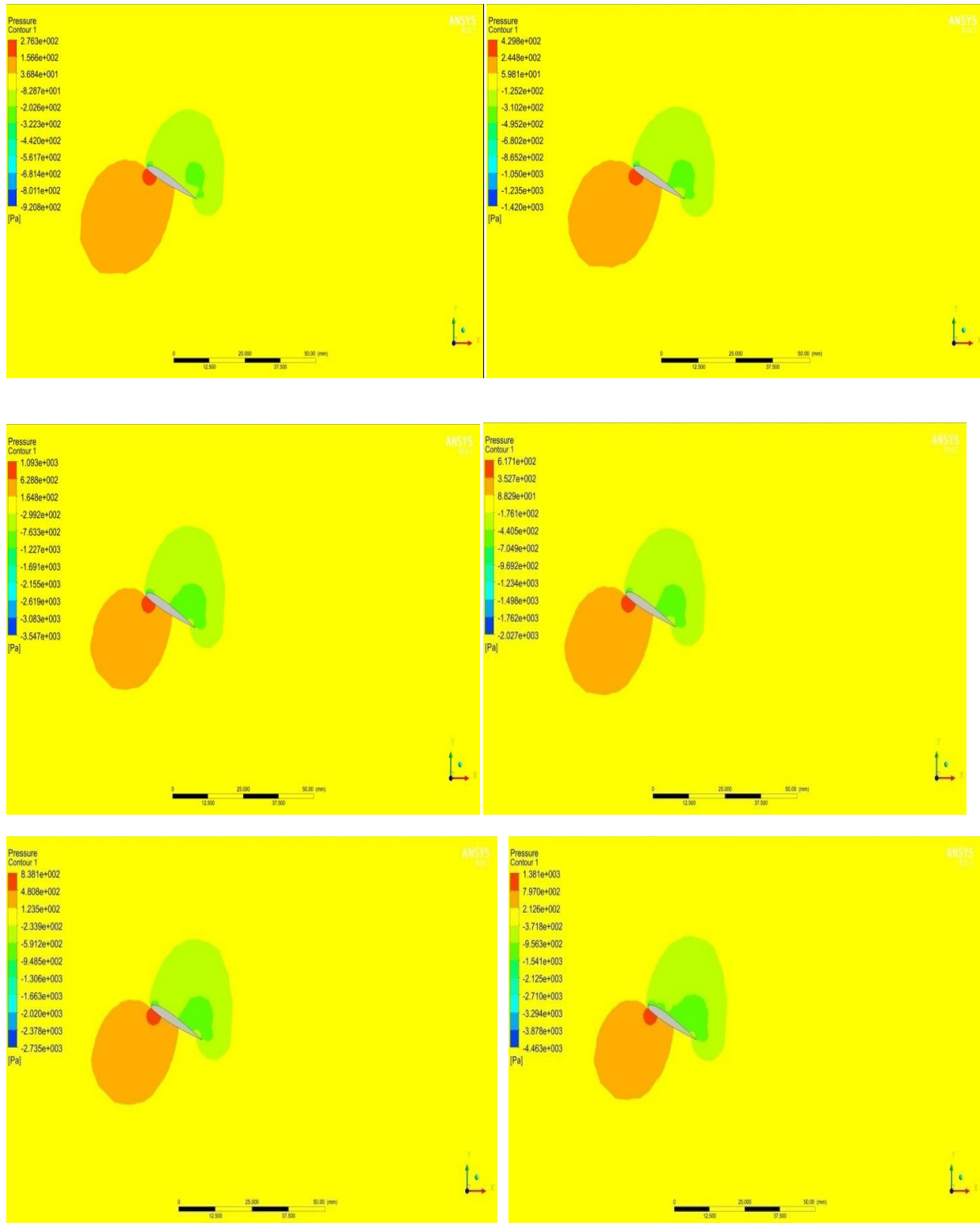


Fig 5-36 Pressure contour at various velocities at 20,25,30,35,40,45 m/sec and 25° AoA for NACA 23012

### 5.1.9 Velocity contour for NACA 23012

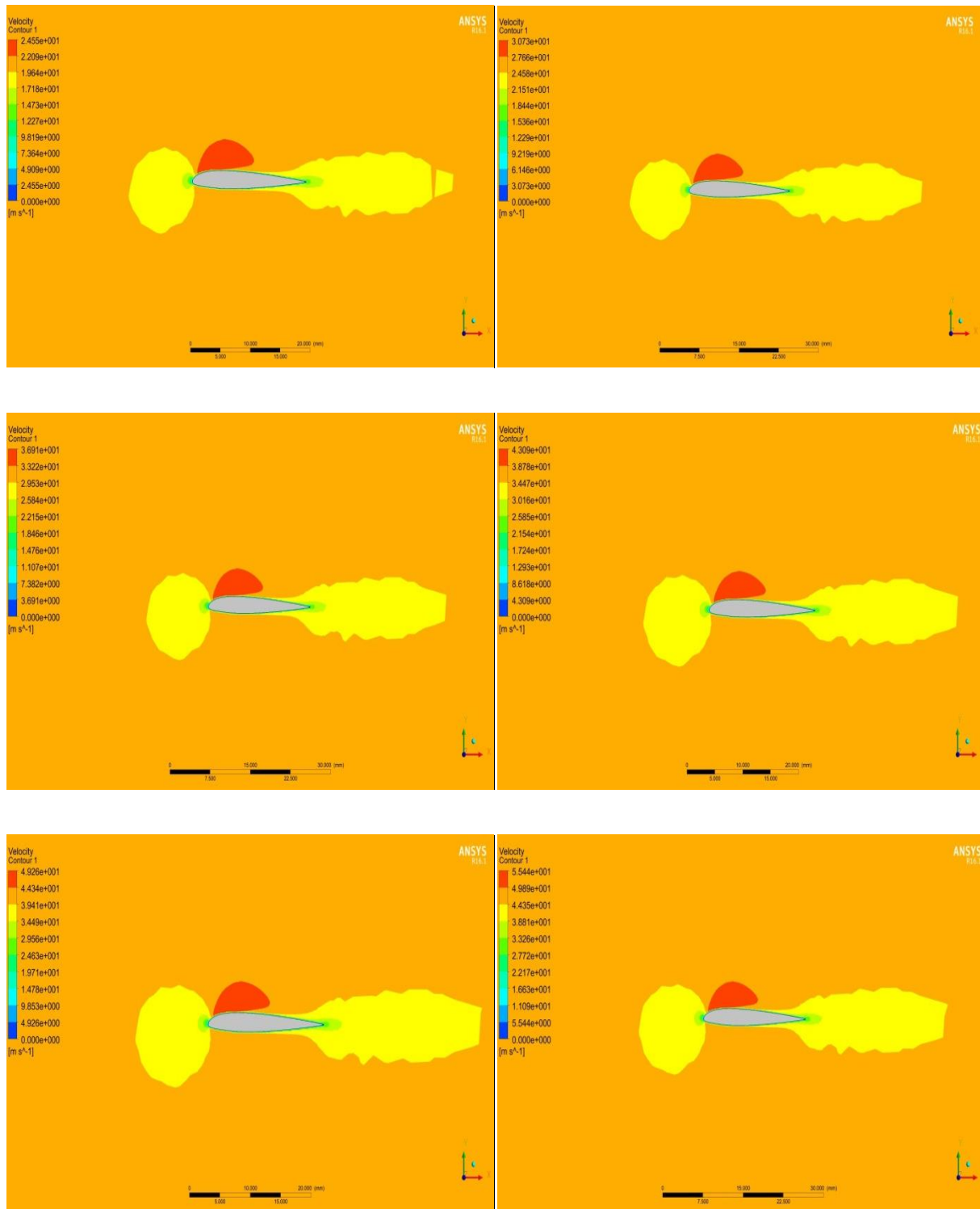


Fig 5-37 Velocity contour plot at various velocities at 20,25,30,35,40,45 m/sec and 0° AoA for NACA 23012

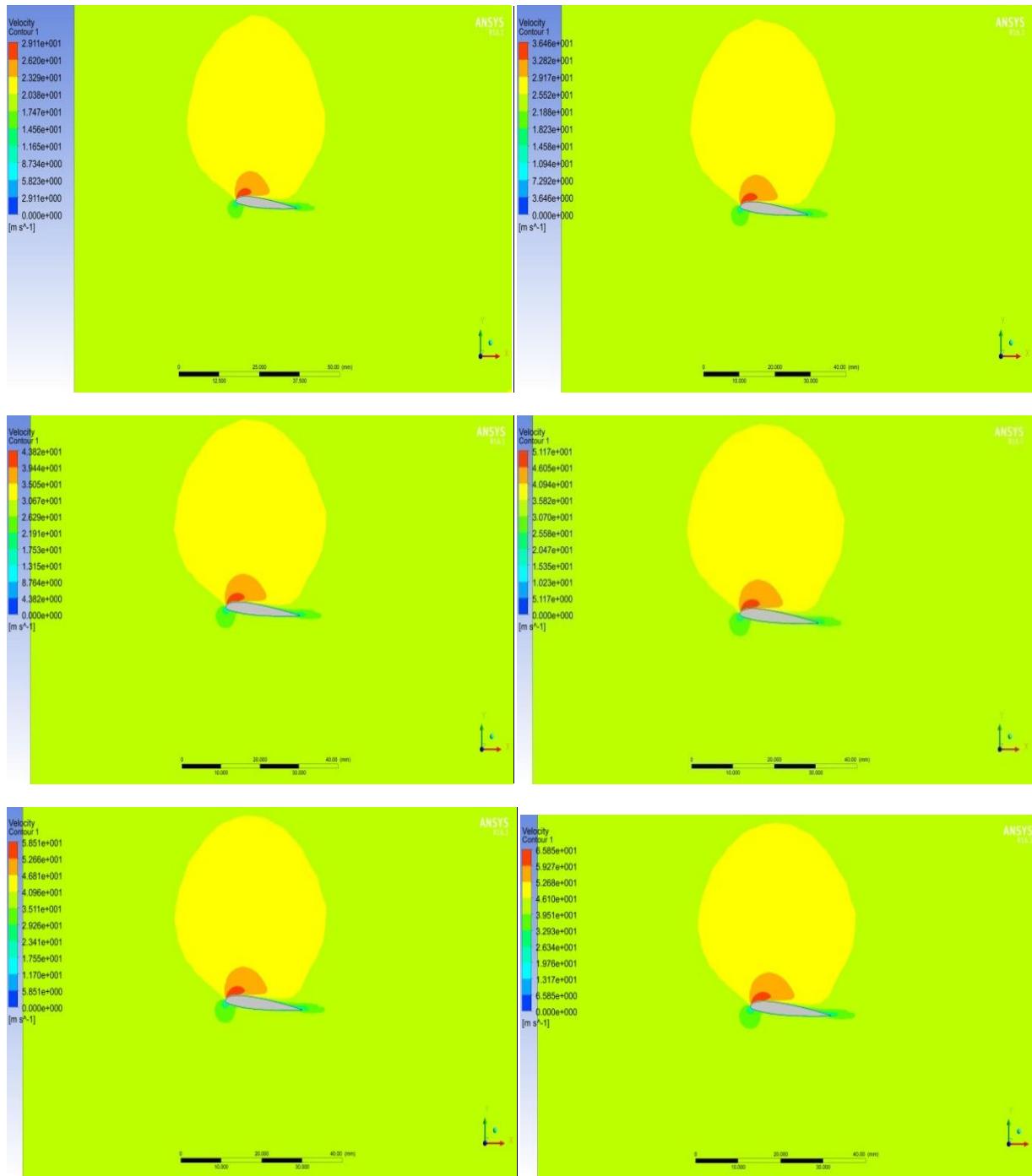


Fig 5-38 Velocity contour plot at various velocities at 20,25,30,35,40,45 m/sec and 5° AoA for NACA 23012

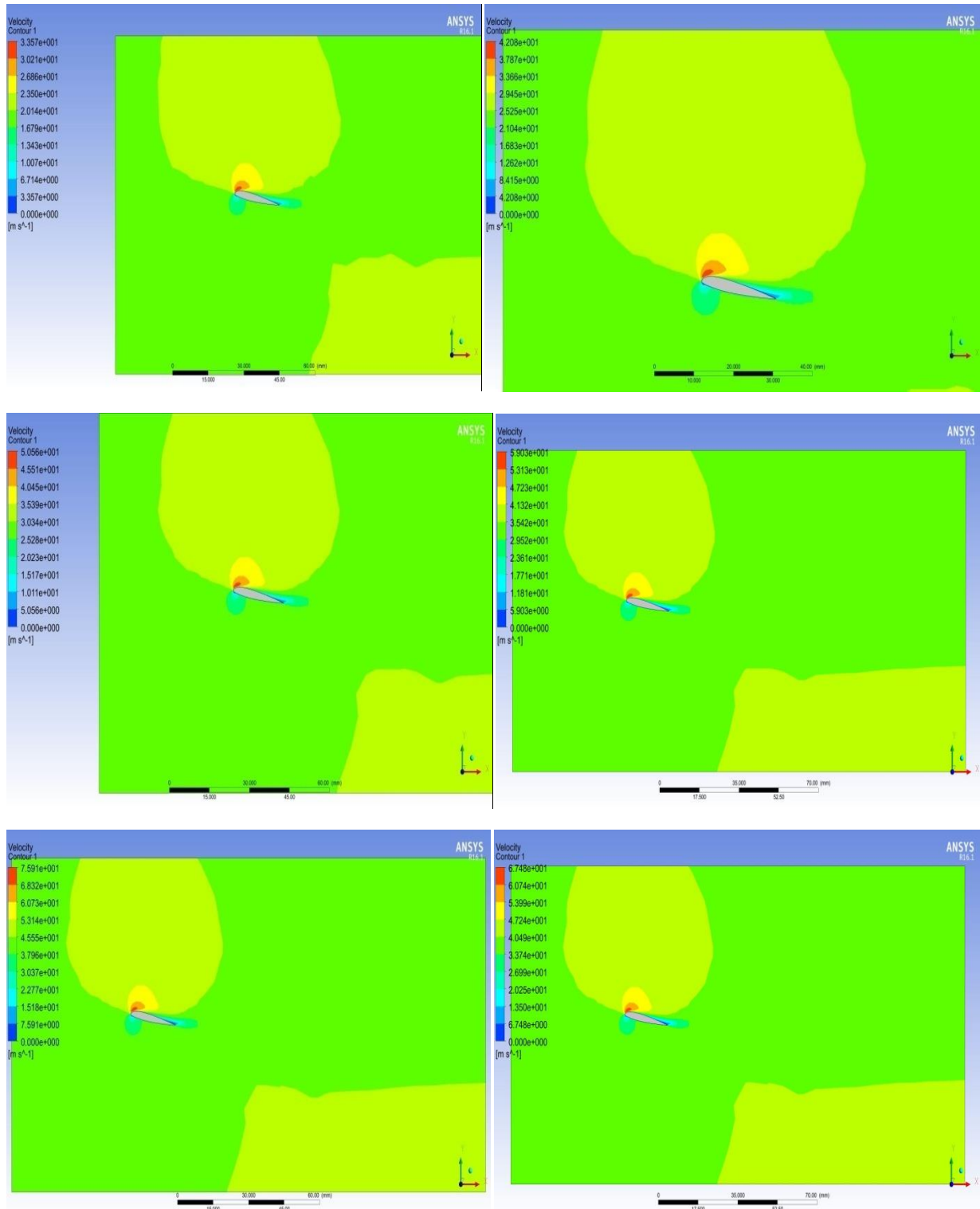


Fig 5-39 Velocity contour plot for various velocities at 20,25,30,35,40,45 m/sec 10° AoA for NACA 23012

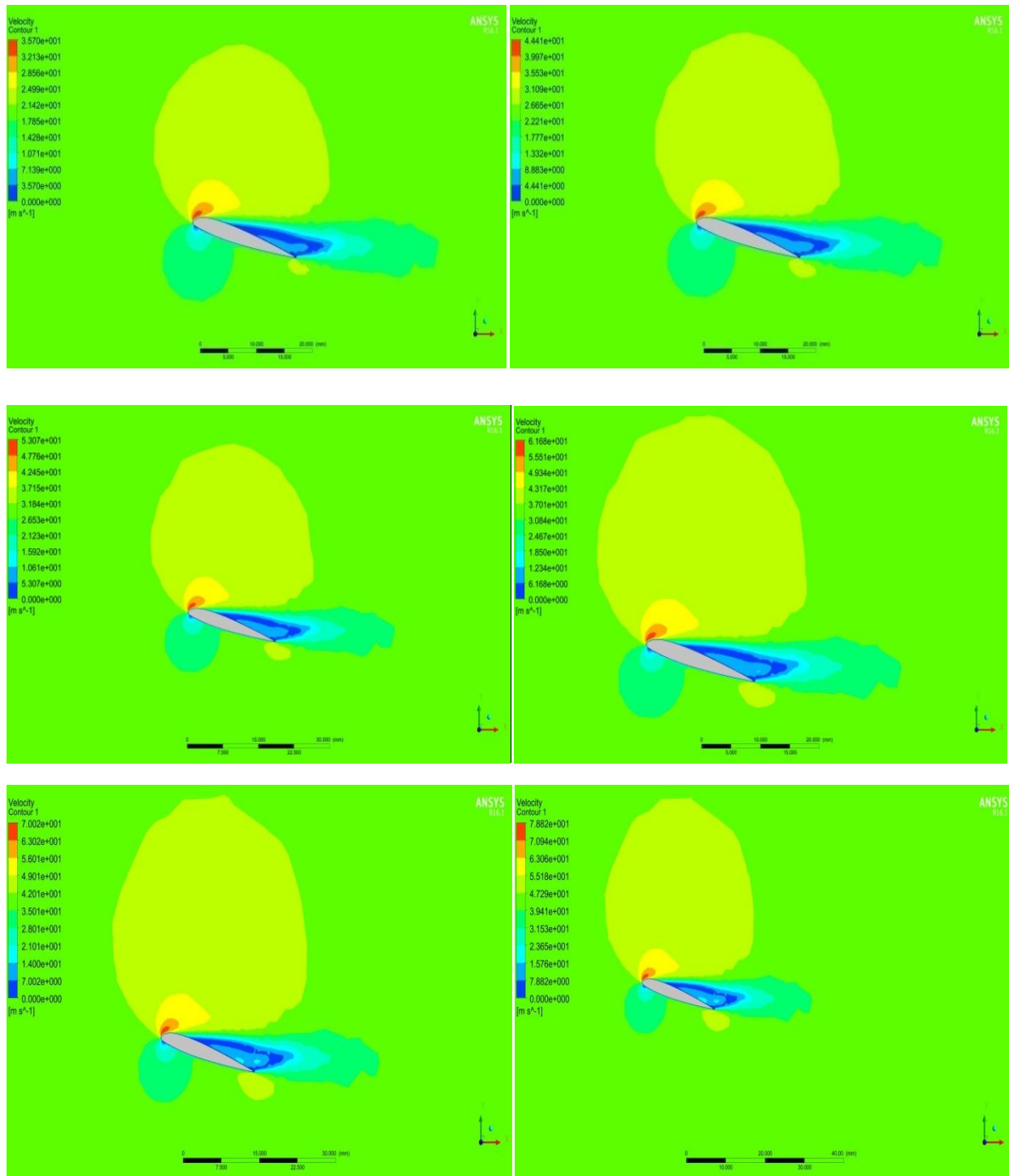


Fig 5-40 Velocity contour plot for various velocities at 20,25,30,35,40,45 m/sec 15° AoA for NACA 23012

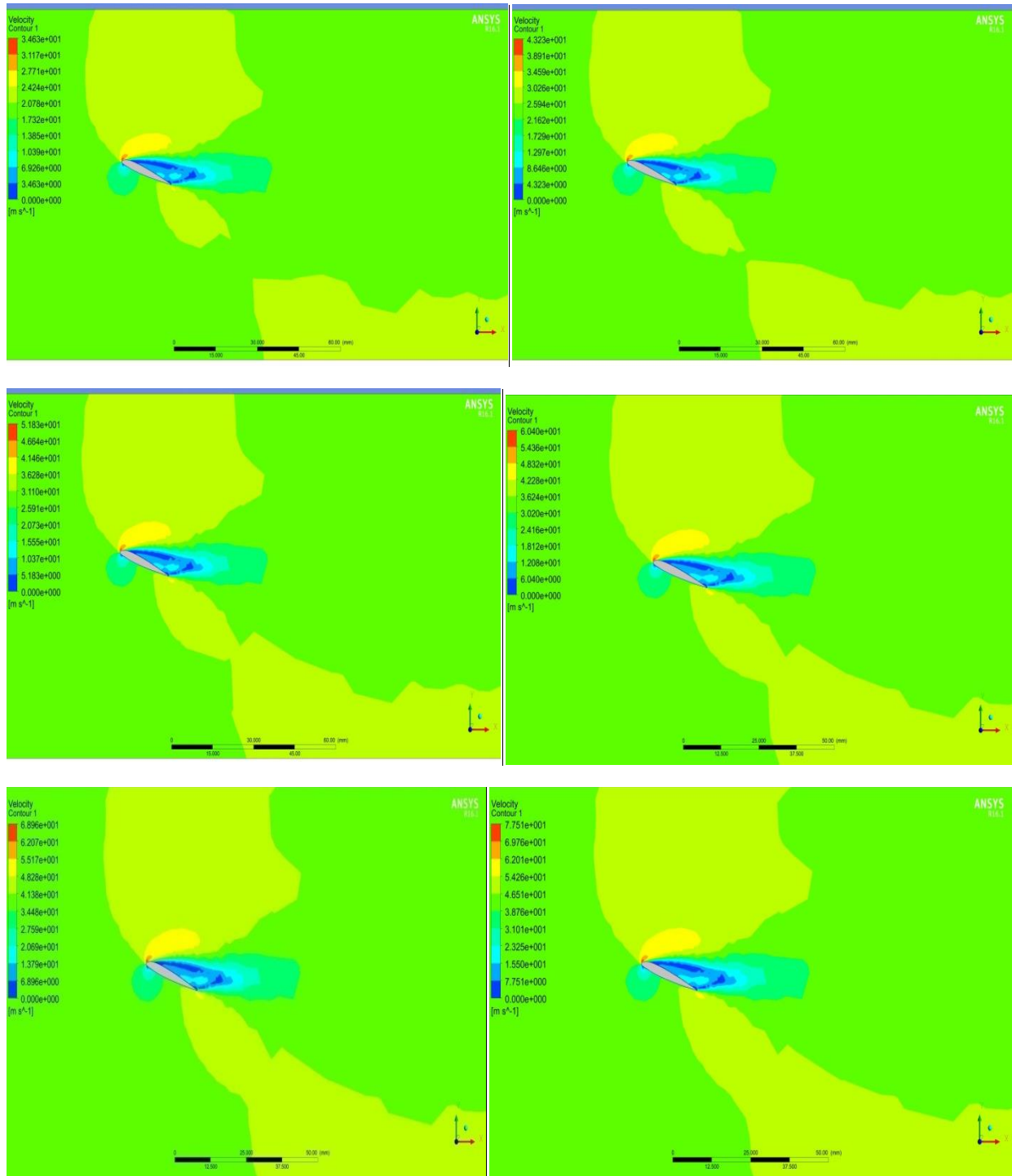


Fig 5-41 Velocity contour plot for various velocities at 20,25,30,35,40,45 m/sec and 20° AoA for NACA 23012

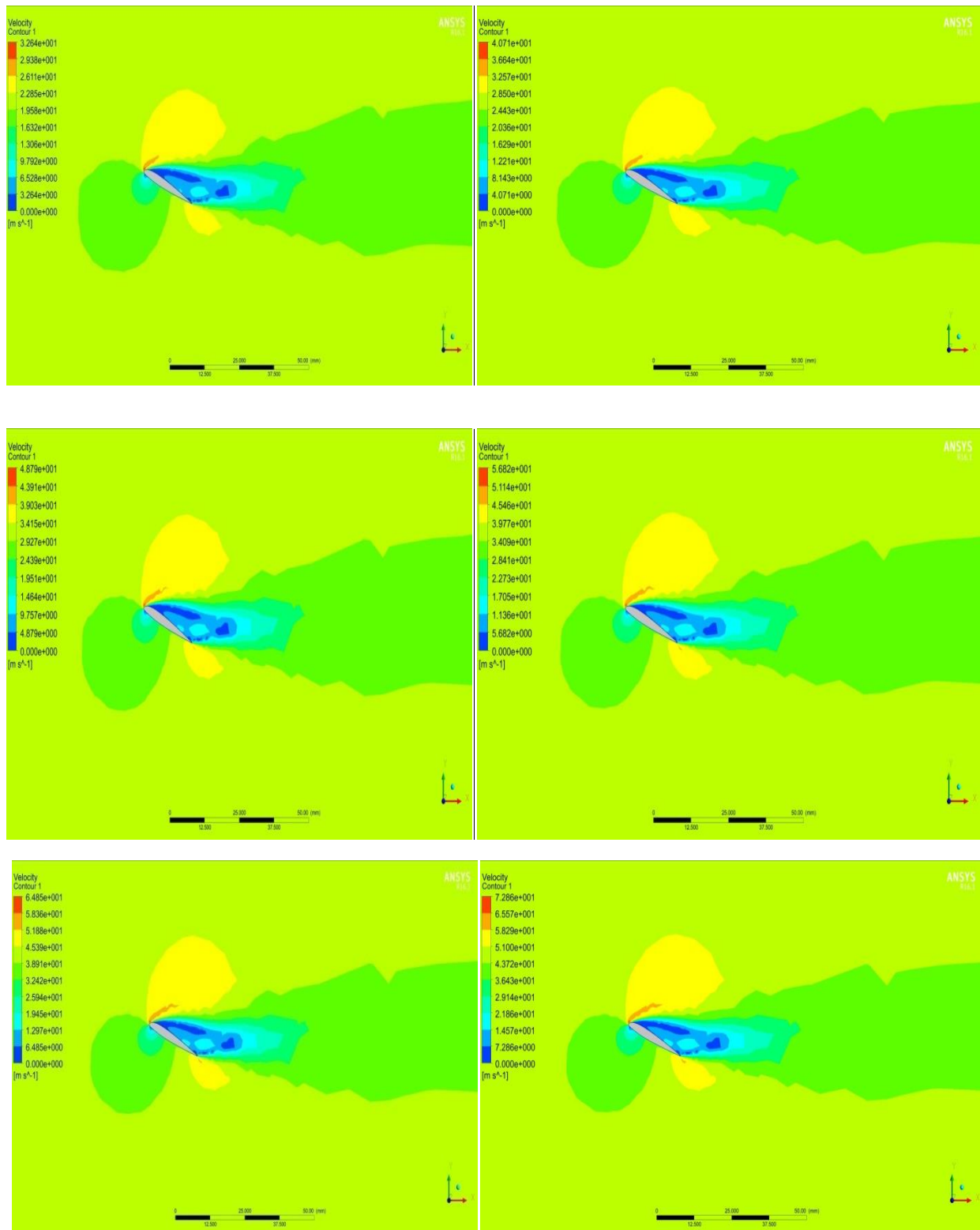


Fig 5-42 Velocity contour plot for various velocities at 20,25,30,35,40,45 m/sec and 25° AoA for NACA 23012

## Pressure contour plotting for variable camber NACA 23012

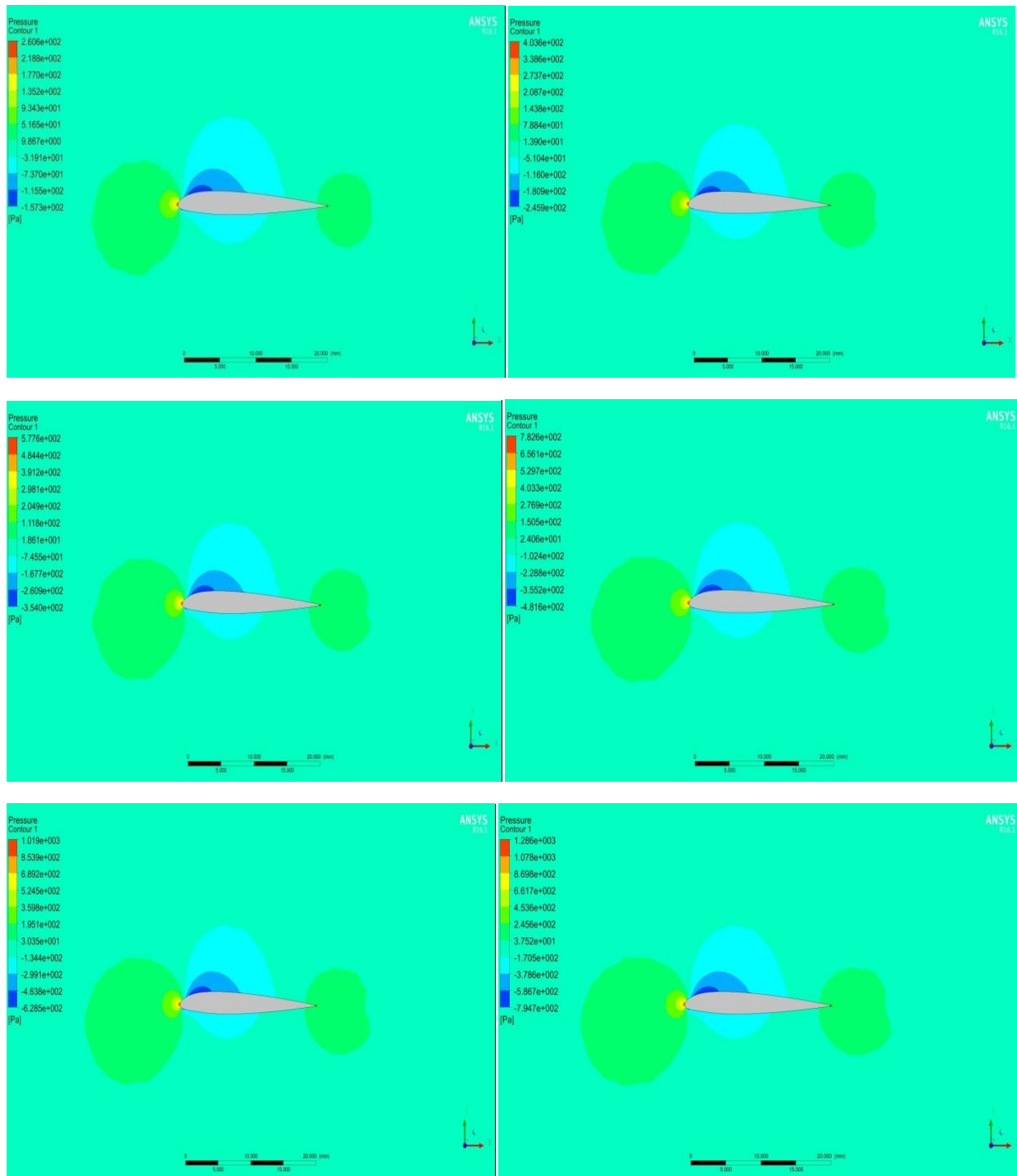


Fig 5-43 Pressure contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 0° AoA for variable camber NACA 23012

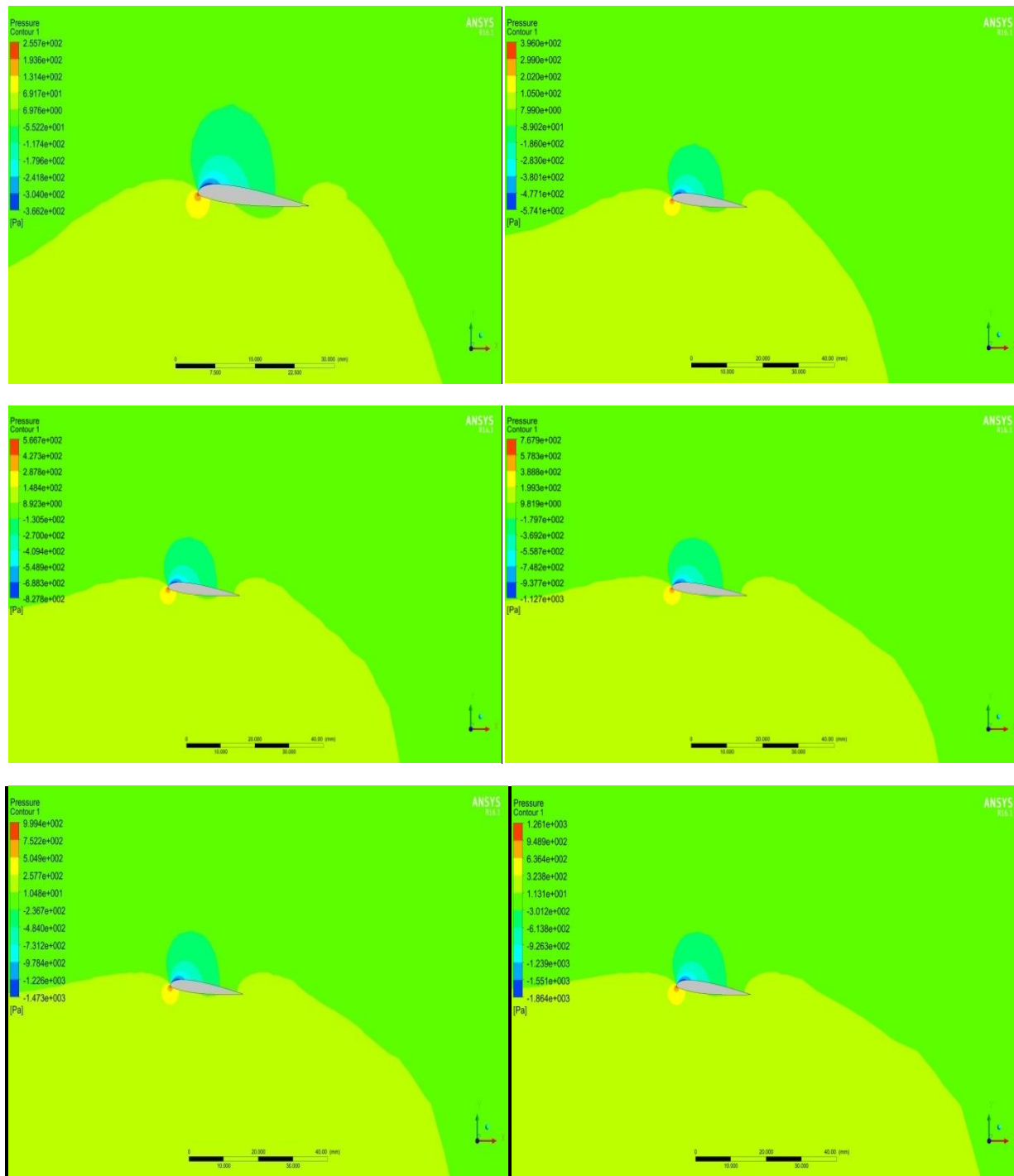


Fig 5-44 Pressure contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 5° AoA for variable camber NACA 23012

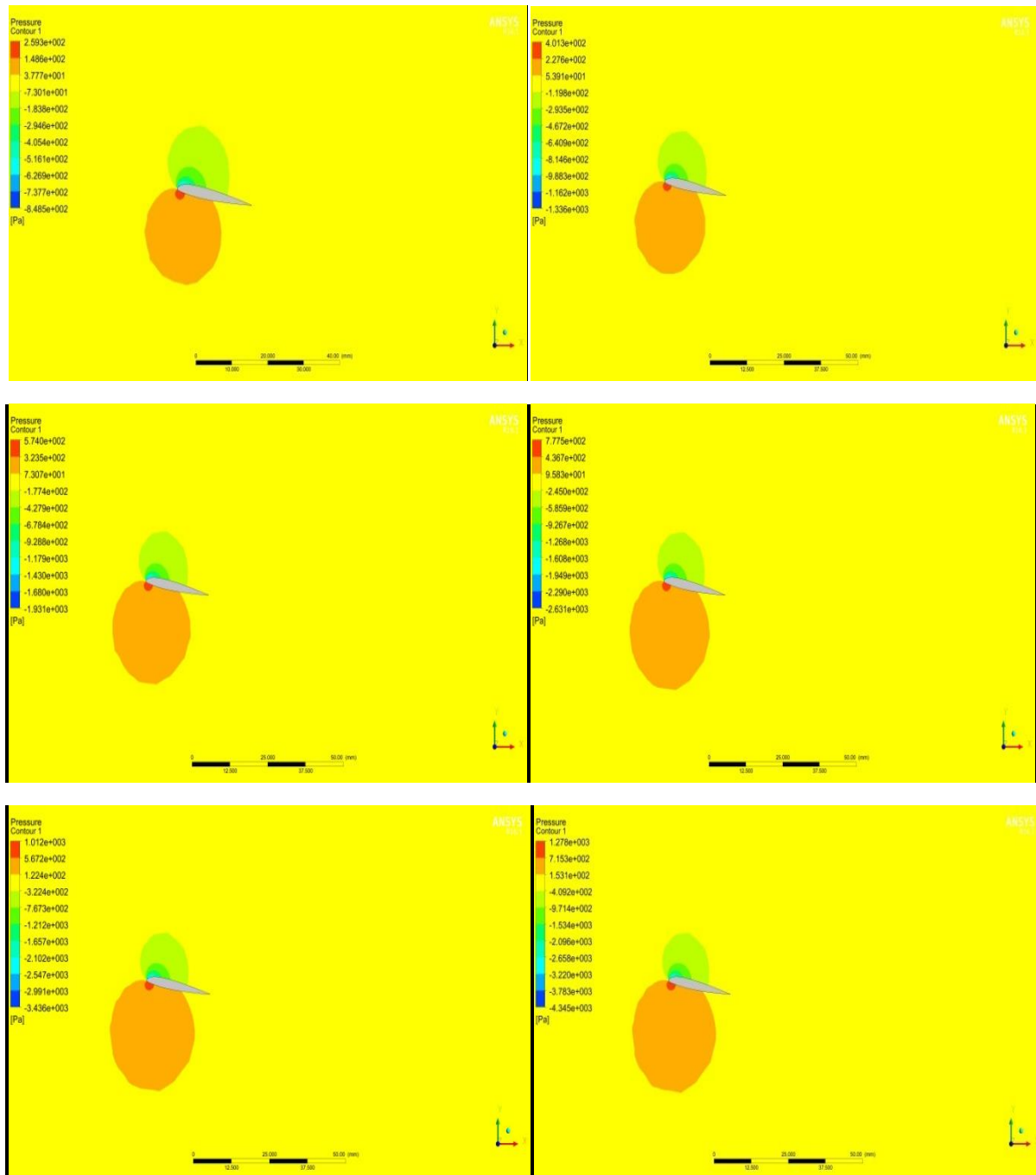


Fig 5-45 Pressure contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 10° AoA for variable camber NACA 23012

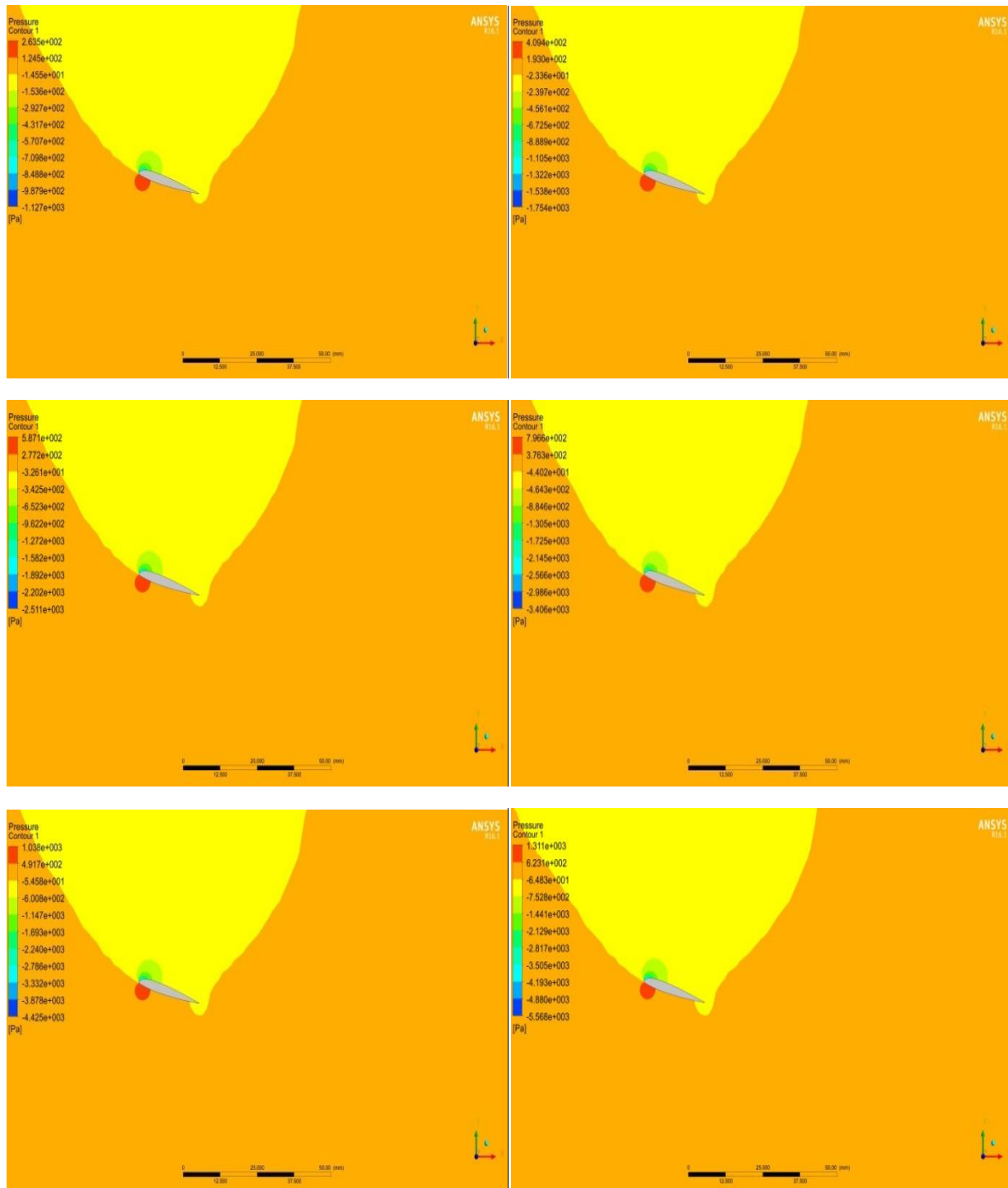


Fig 5-46 Pressure contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 15° AoA for variable camber NACA 23012

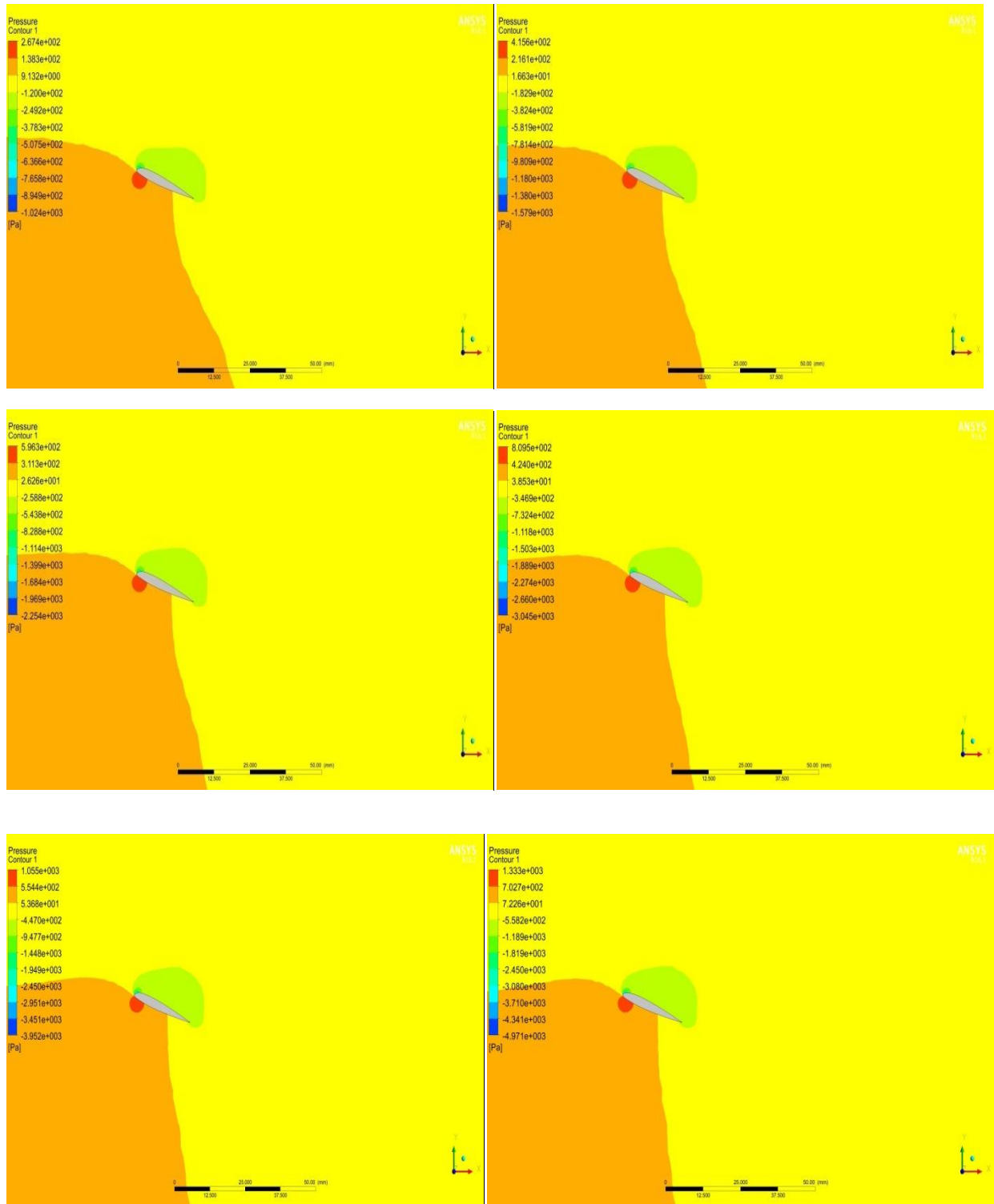


Fig 5-47 Pressure contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 20° AoA for variable camber NACA 23012

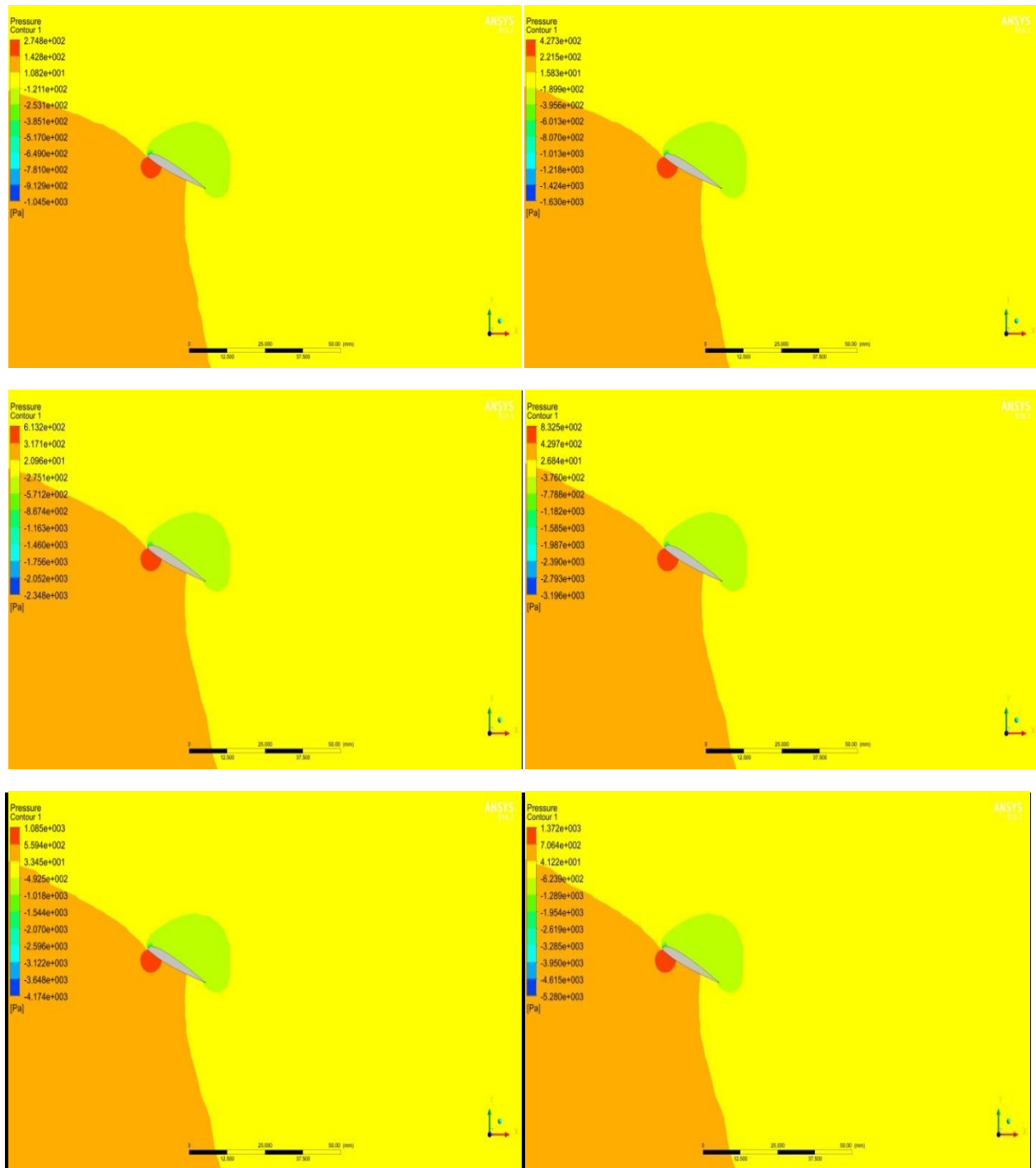


Fig 5-48 Pressure contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 25° AoA for variable camber NACA 23012

## Velocity contour plotting for variable cambered NACA 23012

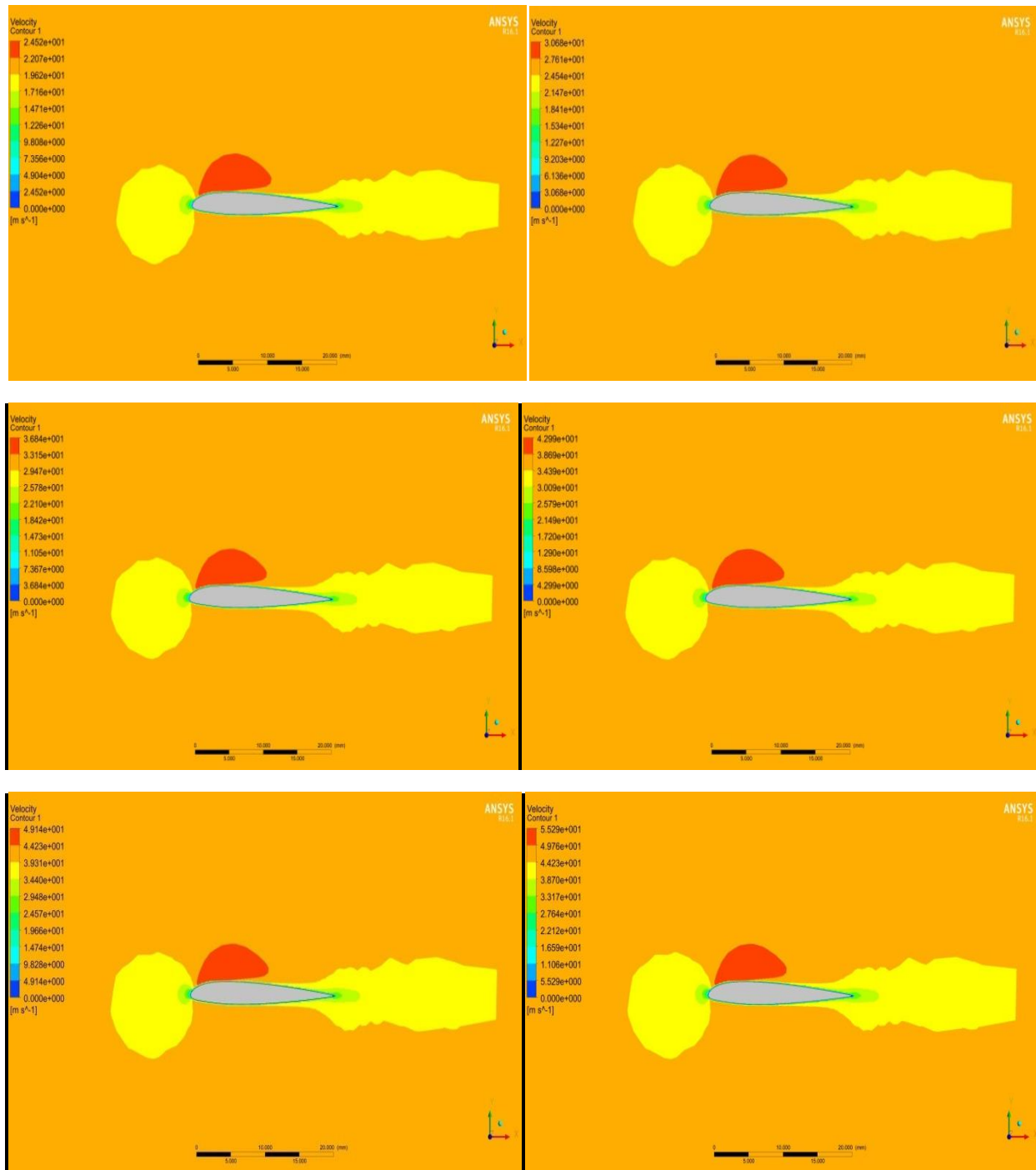


Fig 5-49 Velocity contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 0° AoA for variable camber NACA 23012

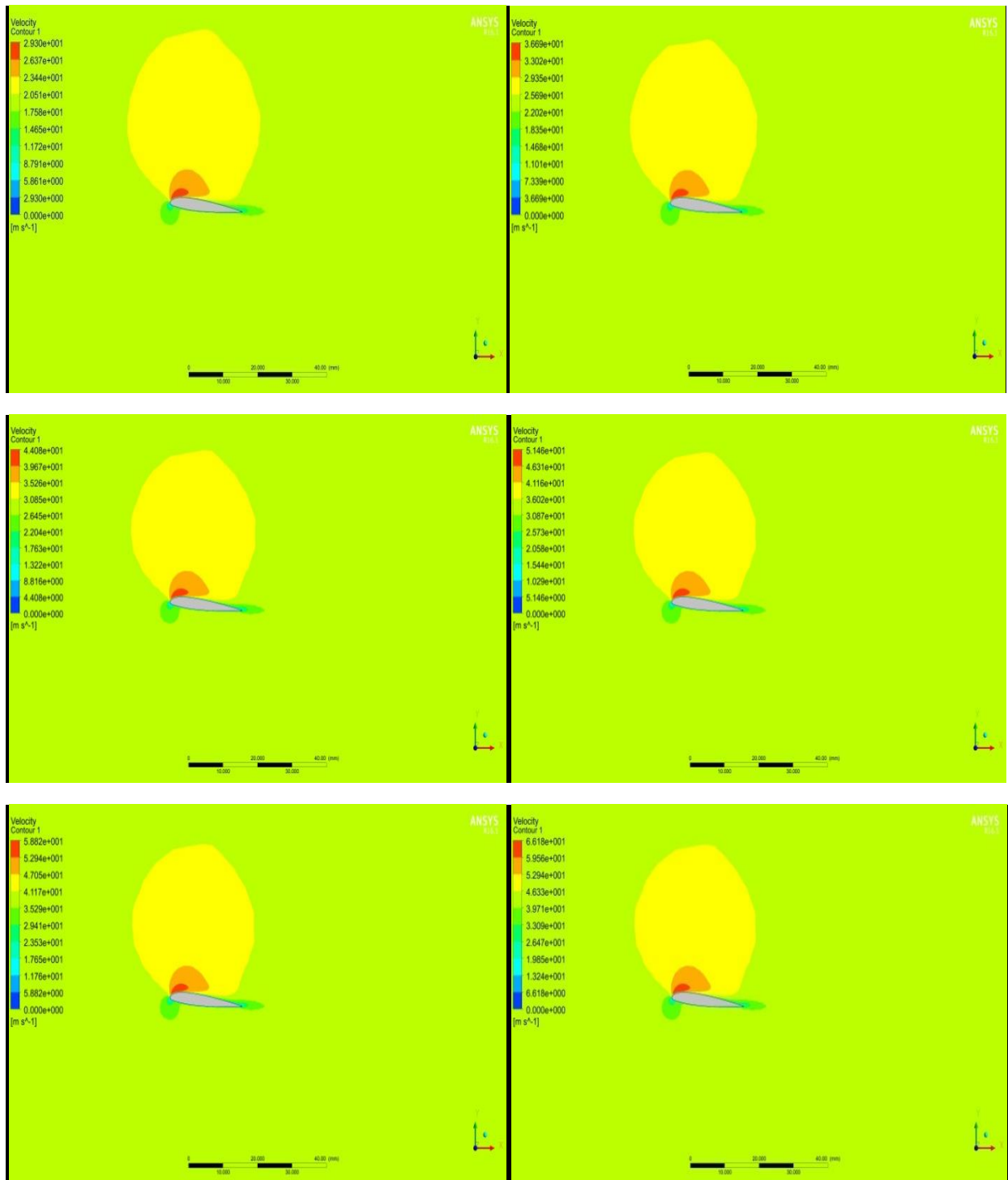


Fig 5-50 Velocity contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 5° AoA for variable camber NACA 23012

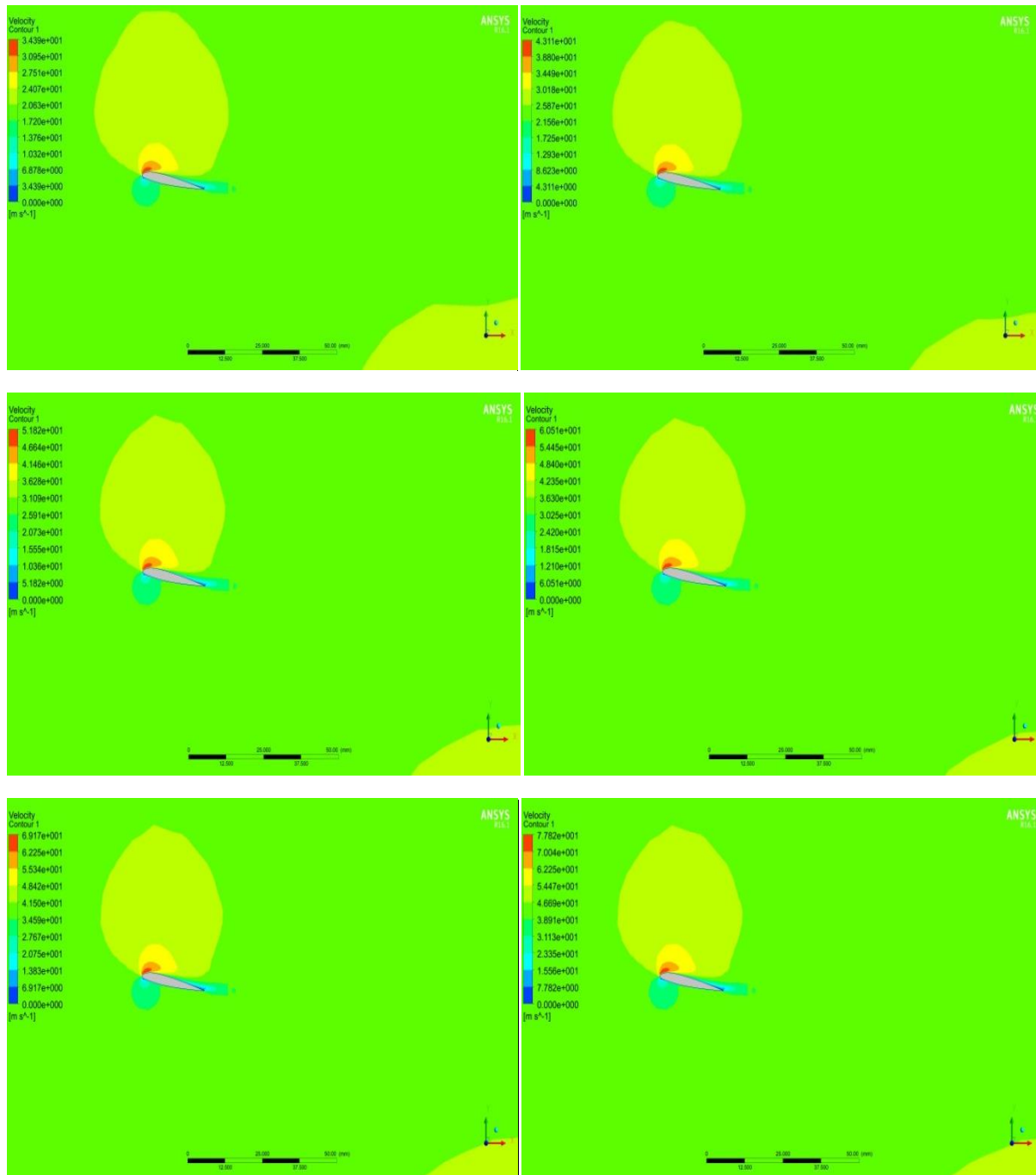


Fig 5-51 Velocity contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 10° AoA  
for variable camber NACA 23012

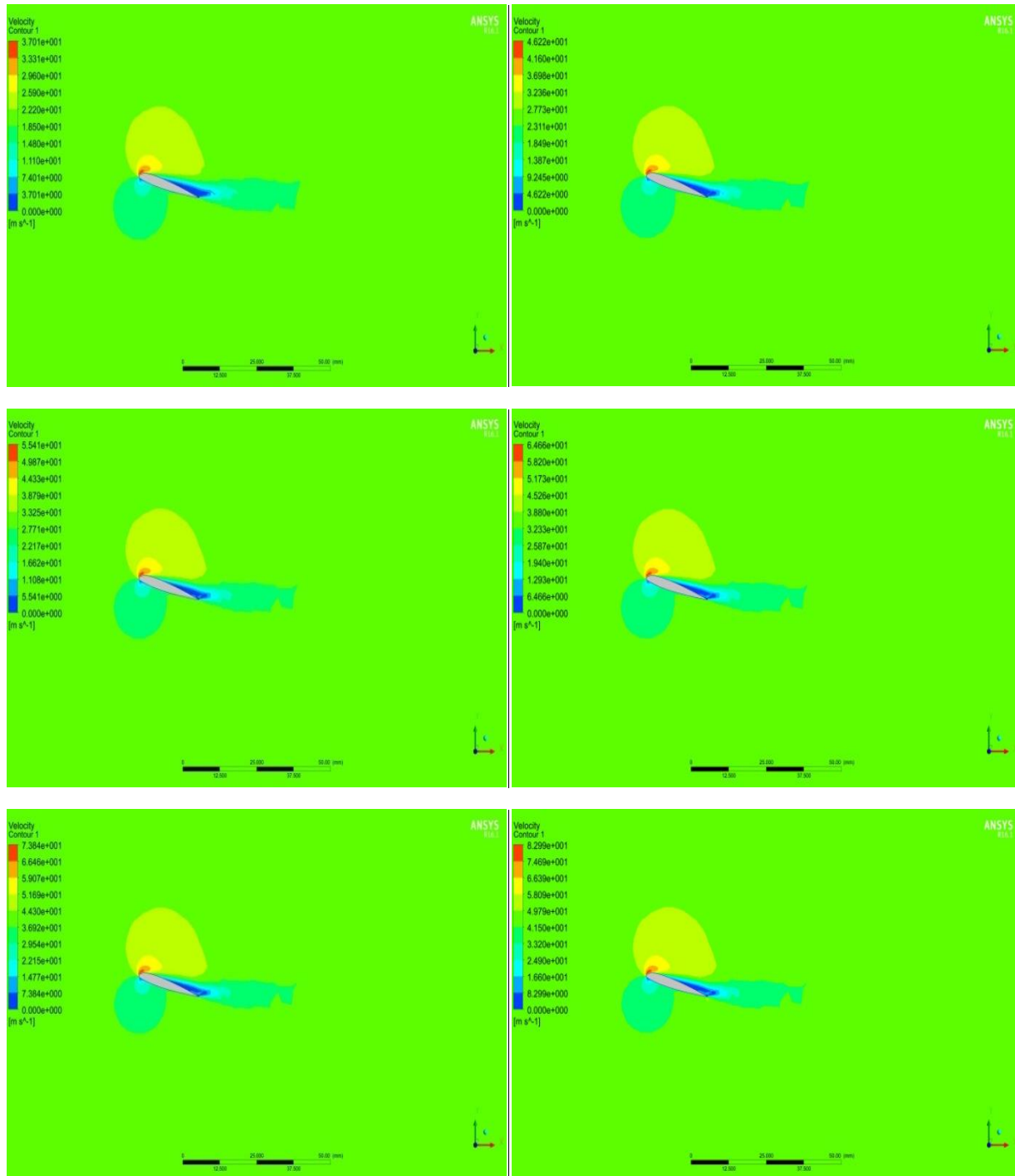


Fig 5-52 Velocity contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 15° AoA  
for variable camber NACA 23012

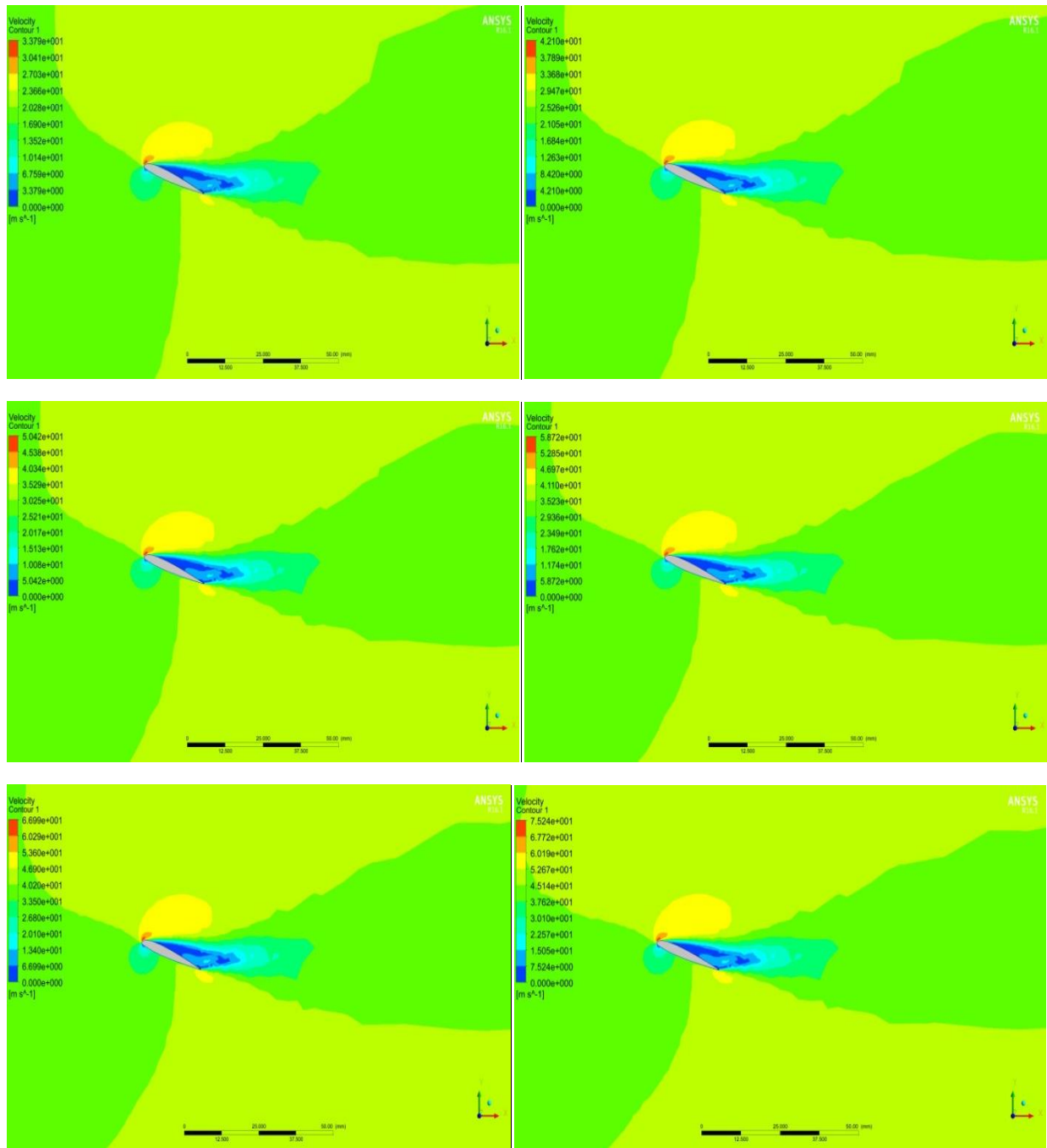


Fig 5-53 Velocity contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 20°  
AoA for variable camber NACA 23012

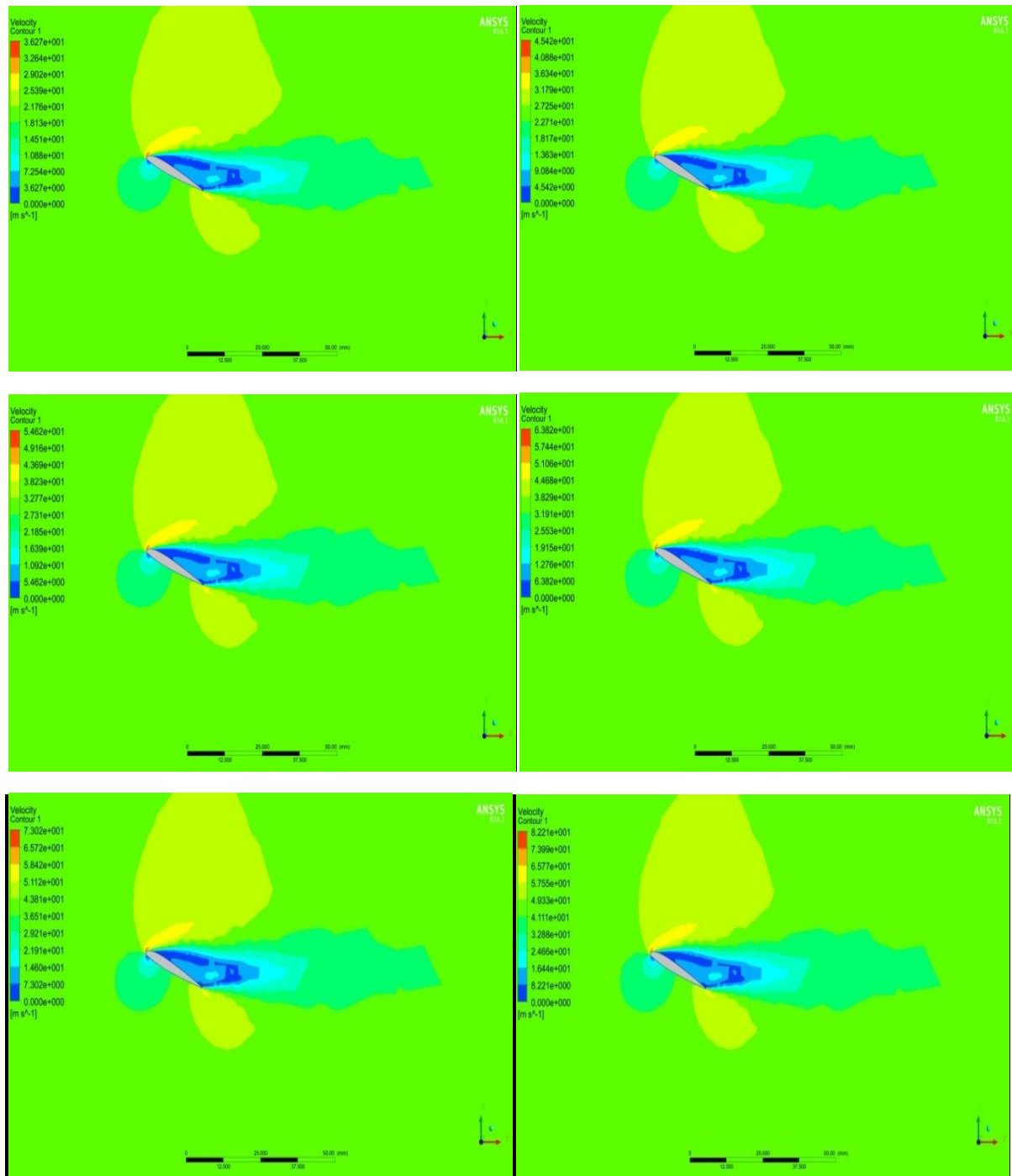


Fig 5-54 Velocity contour plotting for various velocities at 20,25,30,35,40,45 m/sec and 25° AoA  
for variable camber NACA 23012

# CHAPTER 6

## RESULT AND DISCUSSION

## **6. Result and discussions**

In this chapter, Experimental data from the subsonic wind tunnel for NACA 23012 and variable NACA 23012 results that were received are compared with computational, experimental data and graphs are plotted. They will calculate the percentage change in total lift and total drag value and arrived at a conclusion after evaluating the data.

### **6.1 Validation and comparison of theoretical and computational data**

#### **6.1.1 Theoretical data analysis**

The data received from experiments are formulated in the table and are represented below in tabular form; from these data, various plots at different flow velocities are plotted. Lift Vs AoA and Drag Vs AoA are planned, and results are compared.

Plotting  $C_l$  &  $C_d$  at velocity 20 m/sec NACA 23012 and variable camber NACA 23012

| <b>Angle of attack</b> | <b>0°</b> | <b>5°</b> | <b>10°</b> | <b>15°</b> | <b>20°</b> | <b>25°</b> |
|------------------------|-----------|-----------|------------|------------|------------|------------|
| <b>CL 23012</b>        | 0.06406   | 0.328692  | 0.513774   | 0.543677   | 0.518566   | 0.526299   |
| <b>CL Vc 23012</b>     | 0.064734  | 0.357191  | 0.586762   | 0.611834   | 0.553848   | 0.593301   |
| <b>CD 23012</b>        | 0.012488  | 0.017222  | 0.03828    | 0.078666   | 0.146491   | 0.222593   |
| <b>CD Vc 23012</b>     | 0.013226  | 0.017222  | 0.036415   | 0.079225   | 0.160115   | 0.235763   |

Table 6-1  $C_l$  and  $C_d$  value for NACA 23012 and Variable camber NACA 23012 (Vc)

Plotting  $C_l$  and  $C_d$  at velocity 20 m/sec NACA 23012 and variable camber NACA23012

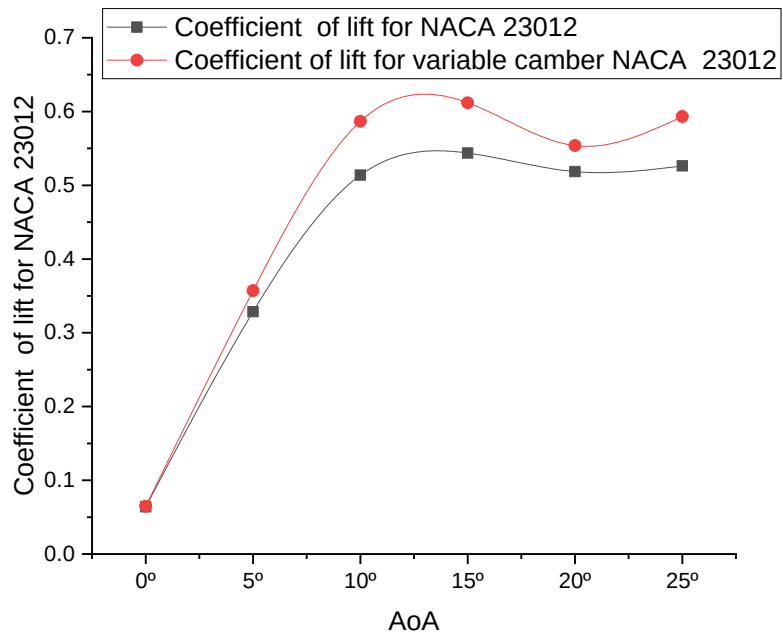


Fig 6-1 Coefficient of lift variation at 20 m/sec

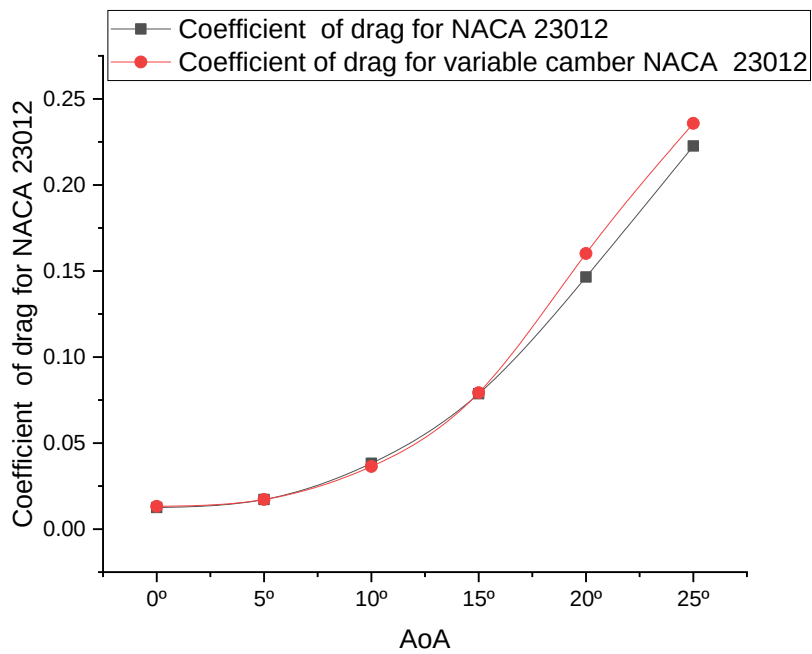


Fig 6-2 Coefficient of drag variation at 20 m/sec

In the above and below graphs we clearly see an increase in Coefficient of lift and Coefficient of drag for experimental data collected from wind tunnel test results variable camber NACA aerfoil when compared with NACA 23012 series at 20 m/sec, 25 m/sec, 30 m/sec, 35 m/sec, 40 m/sec, 45 m/sec, at various angle of attacks. It Indicates that the variable camber NACA 23012 gives better test results when chord length varies for 2cm ie from 19cm to 21 cm, and as expected the results are promising and when the aircraft require lift augmentation this camber morphing technique will increase lift at different flight regimes.

Plotting  $C_L$  and  $C_D$  at velocity 25 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.064075 | 0.329138 | 0.511565 | 0.535337 | 0.513126 | 0.523092 |
| <b>CL Vc 23012</b> | 0.064671 | 0.357558 | 0.586555 | 0.604795 | 0.54892  | 0.590964 |
| <b>CD 23012</b>    | 0.012085 | 0.016673 | 0.037462 | 0.079333 | 0.146359 | 0.222742 |
| <b>CD Vc 23012</b> | 0.012823 | 0.017232 | 0.035354 | 0.079489 | 0.159941 | 0.235644 |

Table 6-2  $C_L$  and  $C_D$  value for NACA 23012 and variable camber NACA 23012 (VC) at 25 m/sec

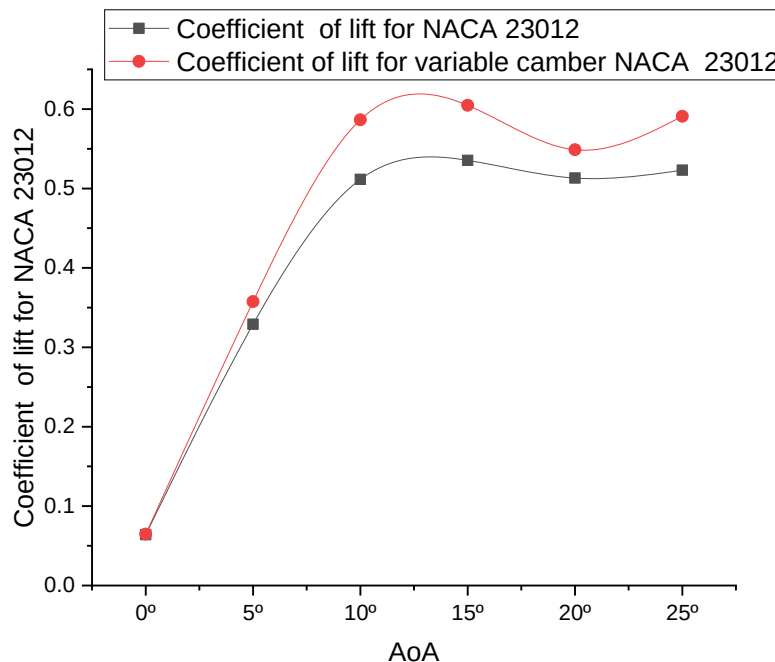


Fig 6-3 Coefficient of lift variation at 25 m/sec

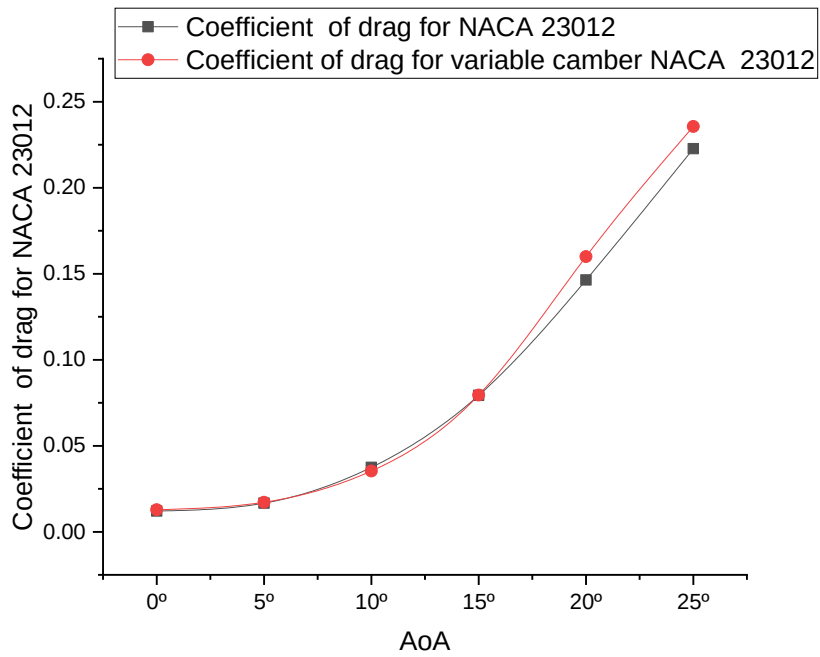


Fig 6-4 Coefficient of drag variation at 25 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 30 m/sec NACA 23012 and variable camber 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.063996 | 0.329148 | 0.509196 | 0.529246 | 0.509352 | 0.521195 |
| <b>CL Vc 23012</b> | 0.064567 | 0.357536 | 0.585686 | 0.598306 | 0.545351 | 0.589451 |
| <b>CD 23012</b>    | 0.011873 | 0.016362 | 0.036953 | 0.079955 | 0.146308 | 0.223003 |
| <b>CD Vc 23012</b> | 0.012624 | 0.016933 | 0.034677 | 0.08002  | 0.159993 | 0.235592 |

Table 6-3  $C_l$  and  $C_D$  value for NACA 23012 and variable camber NACA 23012 at 30 m/sec

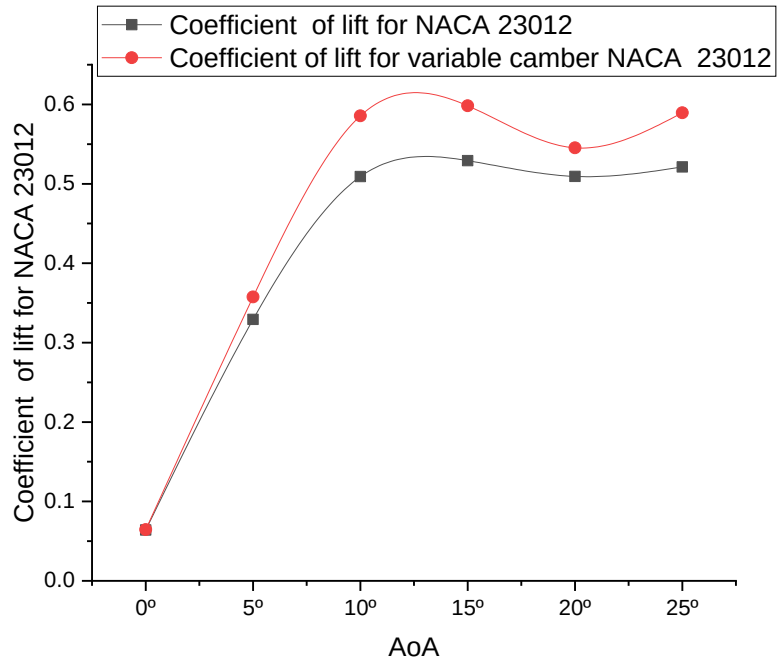


Fig 6-5 Coefficient of lift variation at 30 m/sec

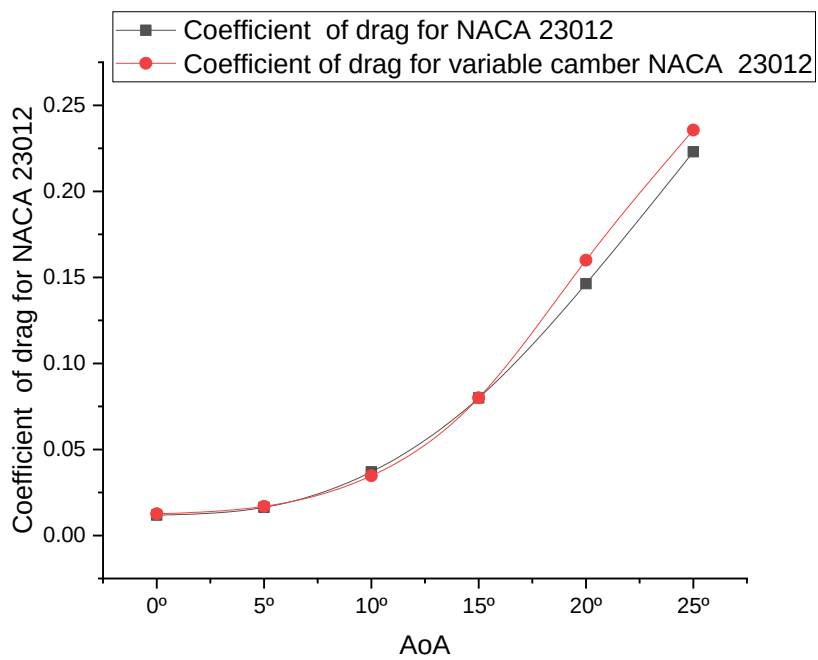


Fig 6-6 Coefficient of drag variation at 30 m/sec

Plotting Cl and Cd at velocity 35 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.063867 | 0.328924 | 0.506789 | 0.524528 | 0.50654  | 0.519384 |
| <b>CL Vc 23012</b> | 0.064451 | 0.357263 | 0.584209 | 0.594126 | 0.542584 | 0.588229 |
| <b>CD 23012</b>    | 0.011782 | 0.016202 | 0.036639 | 0.080667 | 0.146293 | 0.223184 |
| <b>CD Vc 23012</b> | 0.012554 | 0.016794 | 0.034248 | 0.07994  | 0.160096 | 0.235959 |

Table 6-4 Cl and Cd value for NACA 23012 and variable camber NACA 23012 at 35 m/sec

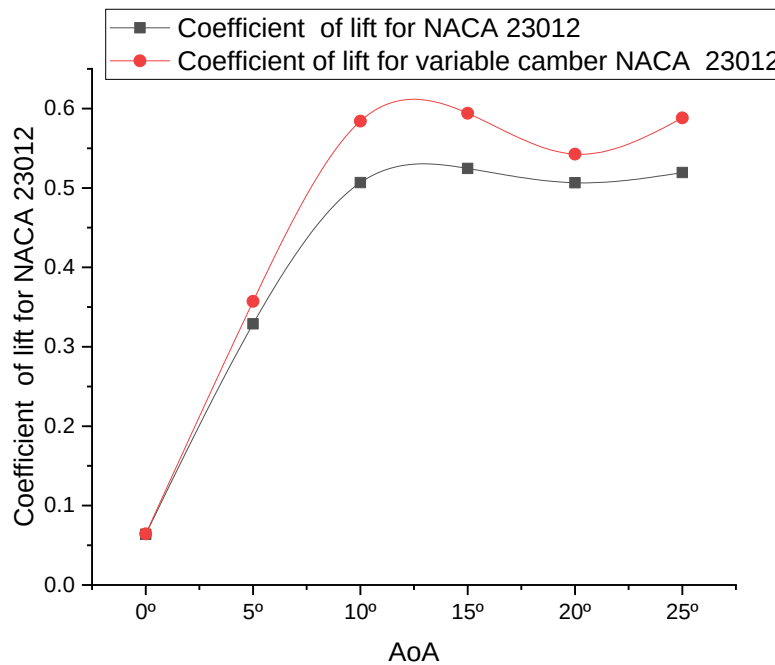


Fig 6-7 coefficient of lift variation at 35 m/sec

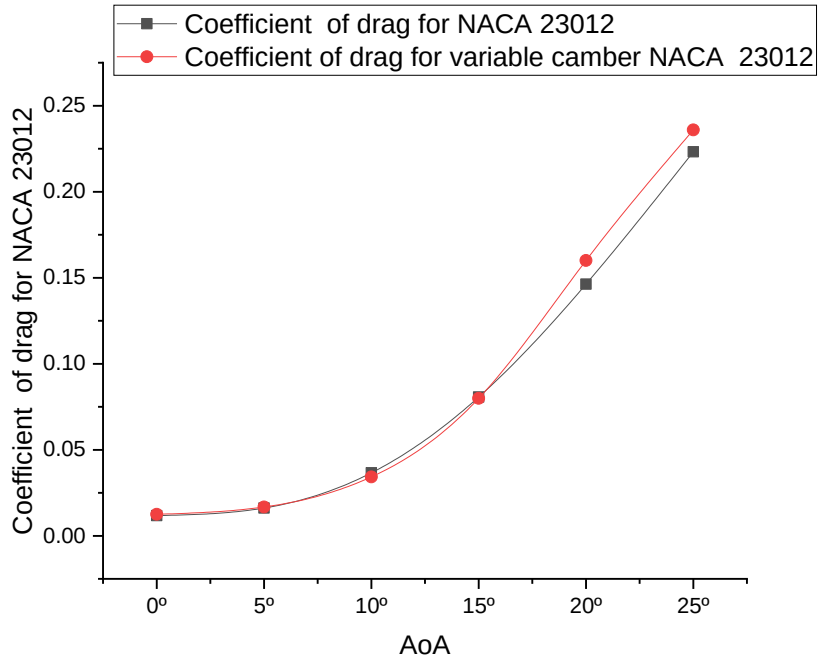


Fig 6-8 Coefficient of drag variation at 35 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 40 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.063715 | 0.328604 | 0.504541 | 0.520079 | 0.504306 | 0.517906 |
| <b>CL Vc 23012</b> | 0.064338 | 0.356847 | 0.582602 | 0.5899   | 0.540321 | 0.587252 |
| <b>CD 23012</b>    | 0.011773 | 0.016141 | 0.03645  | 0.082082 | 0.146317 | 0.223344 |
| <b>CD Vc 23012</b> | 0.012569 | 0.016757 | 0.03398  | 0.080543 | 0.160224 | 0.236095 |

Table 6-5  $C_l$  and  $C_d$  value for NACA 23012 and Variable camber NACA 23012 at 40 m/sec

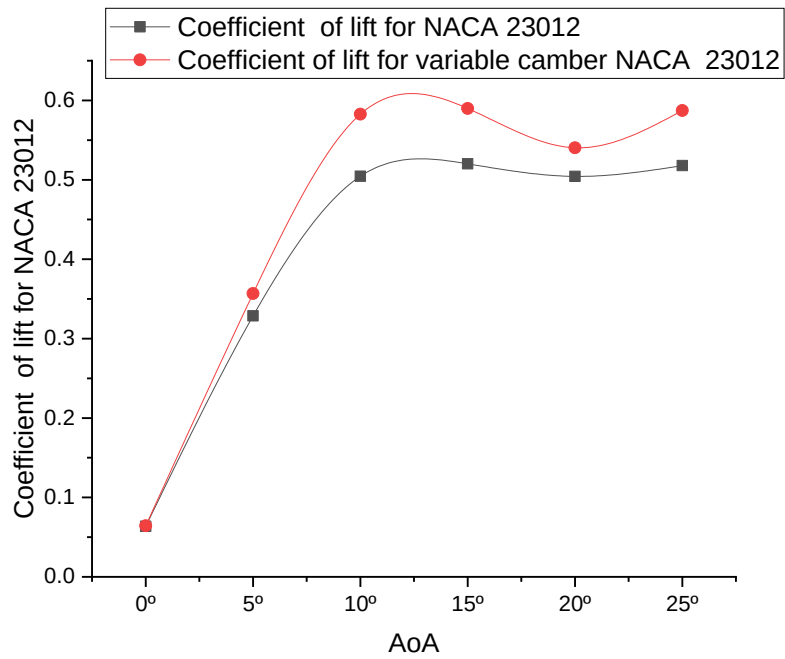


Fig 6-9 Coefficient of lift variation at 40 m/sec

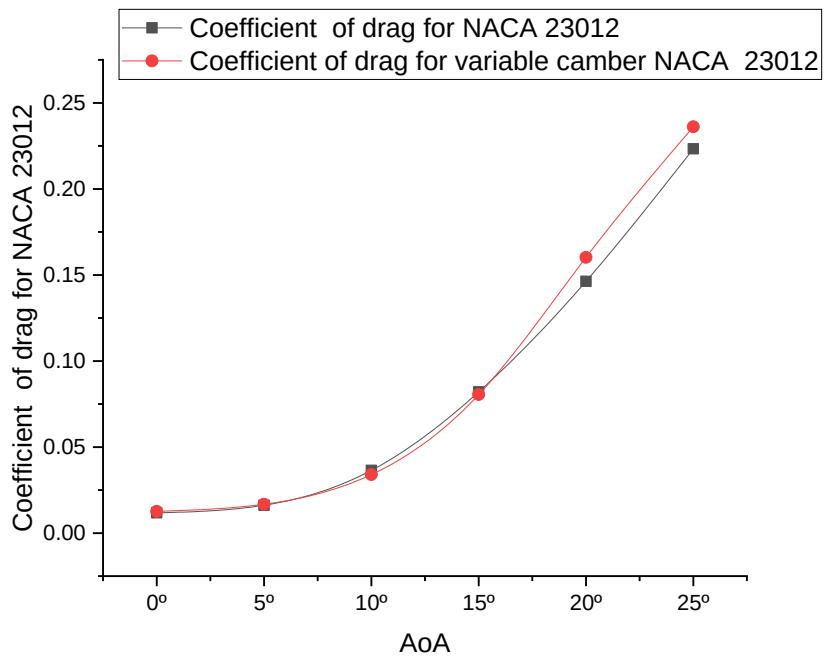


Fig 6-10 Coefficient of drag variation at 40 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 45 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.063544 | 0.328146 | 0.50225  | 0.519321 | 0.502379 | 0.516634 |
| <b>CL Vc 23012</b> | 0.064228 | 0.356396 | 0.580925 | 0.585796 | 0.538396 | 0.586377 |
| <b>CD 23012</b>    | 0.011815 | 0.016146 | 0.036354 | 0.080893 | 0.146369 | 0.223489 |
| <b>CD Vc 23012</b> | 0.012636 | 0.016785 | 0.033822 | 0.080567 | 0.160344 | 0.236244 |

Table 6-6  $C_l$  and  $C_d$  values for NACA 23012 and variable camber NACA 23012 at 45 m/sec

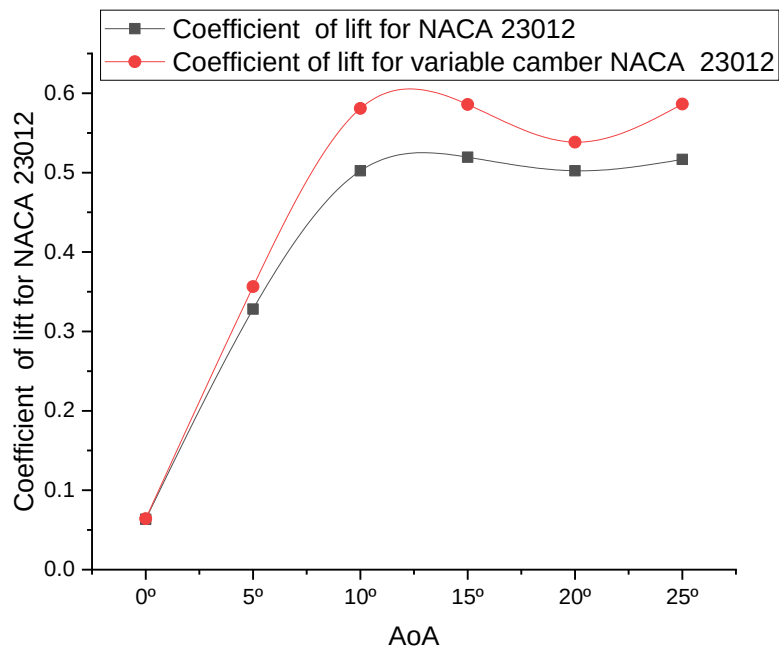


Fig 6-11 Coefficient of lift variation at 45 m/sec

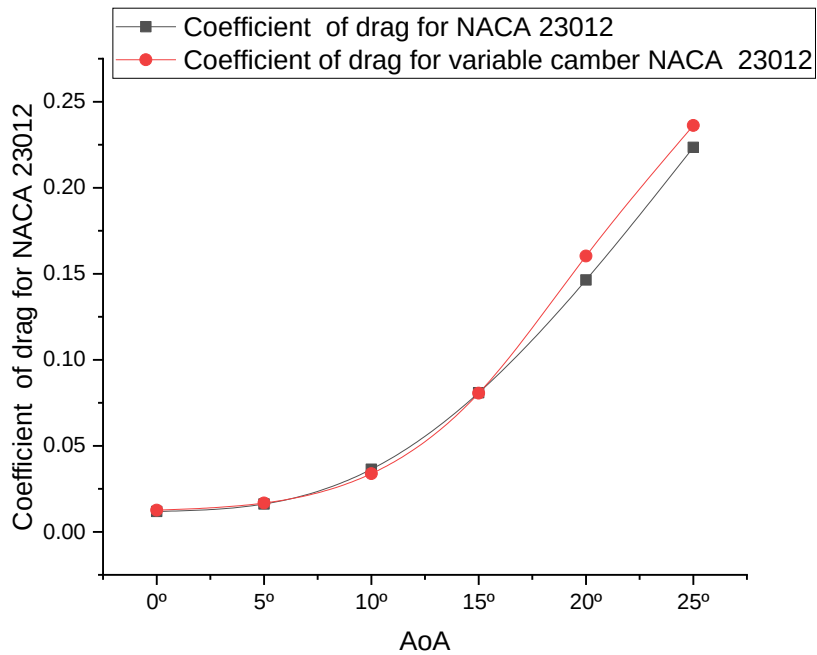


Fig 6-12 Coefficient of drag variation at 45 m/sec

### 6.1.2 Computational data analysis

In the below graphs we clearly see an increase in Coefficient of lift and Coefficient of drag for computational data collected from wind tunnel test results variable camber NACA aerofoil when compared with NACA 23012 series at 20 m/sec, 25 m/sec, 30 m/sec, 35 m/sec, 40 m/sec, 45 m/sec, at various angle of attacks. It indicates that the variable camber NACA 23012 gives better test results when chord length varies for 2 cm i.e. from 19 cm to 21 cm, and as expected the results are promising and when the aircraft requires lift augmentation this camber morphing technique will increase lift at different flight regimes.

Plotting  $C_L$  and  $C_d$  at velocity 20 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.061597 | 0.31605  | 0.494013 | 0.522766 | 0.498621 | 0.506057 |
| <b>CL Vc 23012</b> | 0.062244 | 0.343453 | 0.564194 | 0.588302 | 0.532546 | 0.570482 |
| <b>CD 23012</b>    | 0.012007 | 0.01656  | 0.036808 | 0.075641 | 0.140857 | 0.214032 |
| <b>CD Vc 23012</b> | 0.012717 | 0.017102 | 0.035015 | 0.076178 | 0.153957 | 0.226695 |

Table 6-7  $C_L$  and  $C_d$  value for NACA 23012 and variable camber NACA 23012 at 20 m/sec

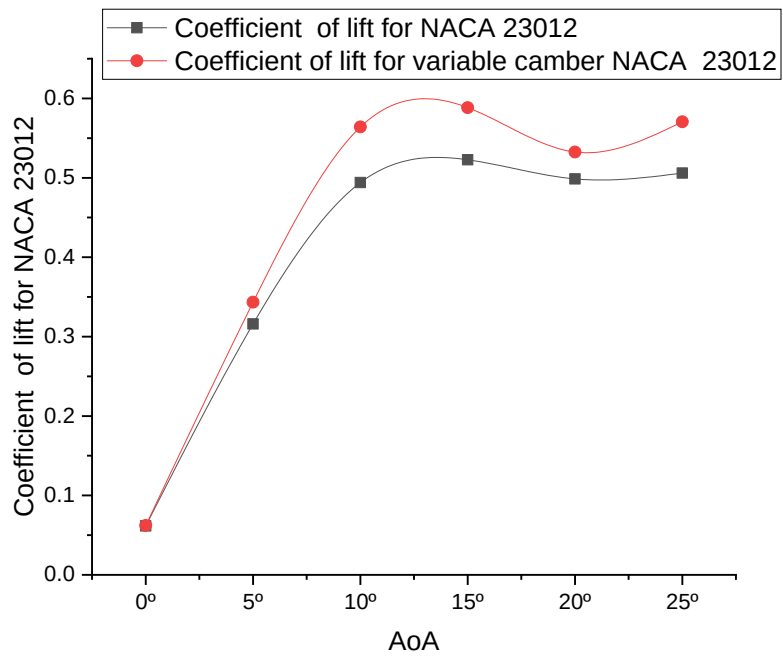


Fig 6-13 Coefficient of lift variation at 20 m/sec

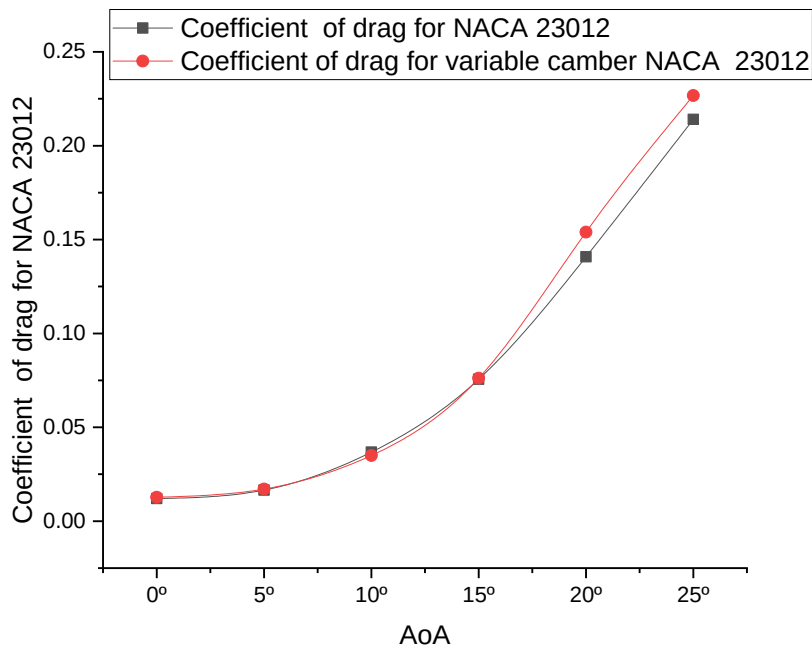


Fig 6-14 Coefficient of drag variation at 20 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 25 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.06161  | 0.316479 | 0.491889 | 0.514747 | 0.49339  | 0.502973 |
| <b>CL Vc 23012</b> | 0.062184 | 0.343806 | 0.563995 | 0.581534 | 0.527808 | 0.568235 |
| <b>CD 23012</b>    | 0.01162  | 0.016032 | 0.036021 | 0.076282 | 0.14073  | 0.214175 |
| <b>CD Vc 23012</b> | 0.01233  | 0.016569 | 0.033994 | 0.076431 | 0.153789 | 0.226581 |

Table 6-8  $C_l$  and  $C_d$  value for NACA 23012 and variable camber NACA 23012 at 25 m/sec

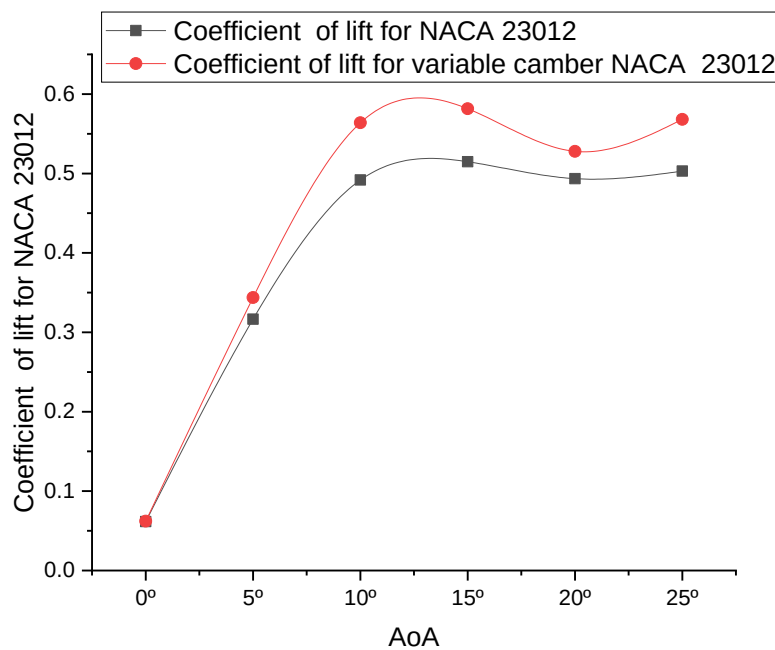


Fig 6-15 Coefficient of lift variation at 25 m/sec

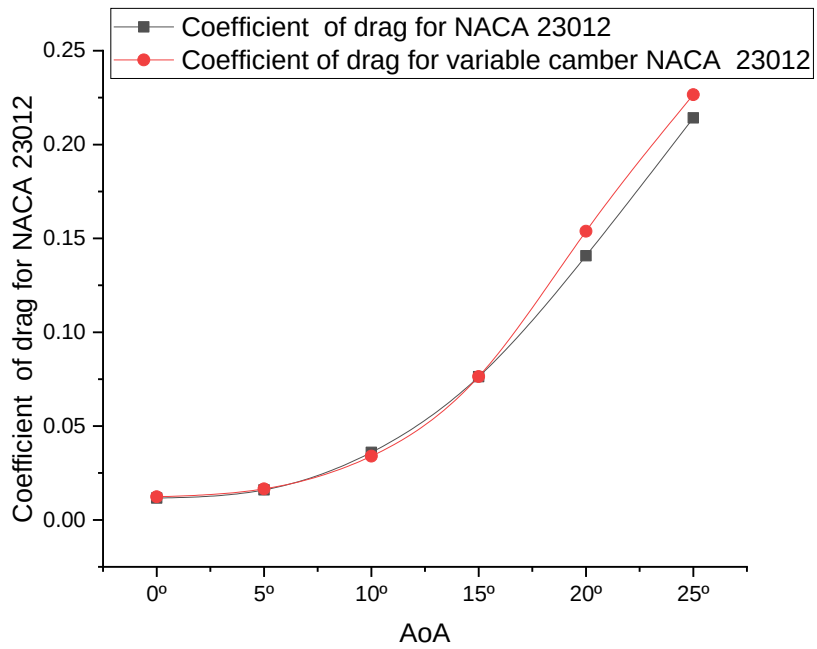


Fig 6-16 Coefficient of drag variation at 25 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 30 m/sec NACA 23012 and variable camber NACA 23012

|                    | 0° AoA   | 5° AoA   | 10° AoA  | 15° AoA  | 20° AoA  | 25° AoA  |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.061535 | 0.316488 | 0.489612 | 0.50889  | 0.489762 | 0.501149 |
| <b>CL Vc 23012</b> | 0.062083 | 0.343785 | 0.56316  | 0.575294 | 0.524376 | 0.56678  |
| <b>CD 23012</b>    | 0.011416 | 0.015733 | 0.035532 | 0.07688  | 0.140681 | 0.214426 |
| <b>CD Vc 23012</b> | 0.012138 | 0.016282 | 0.033343 | 0.076942 | 0.153839 | 0.226531 |

Table 6-9  $C_l$  and  $C_d$  for NACA 23012 and variable camber NACA 23012 at 30 m/sec

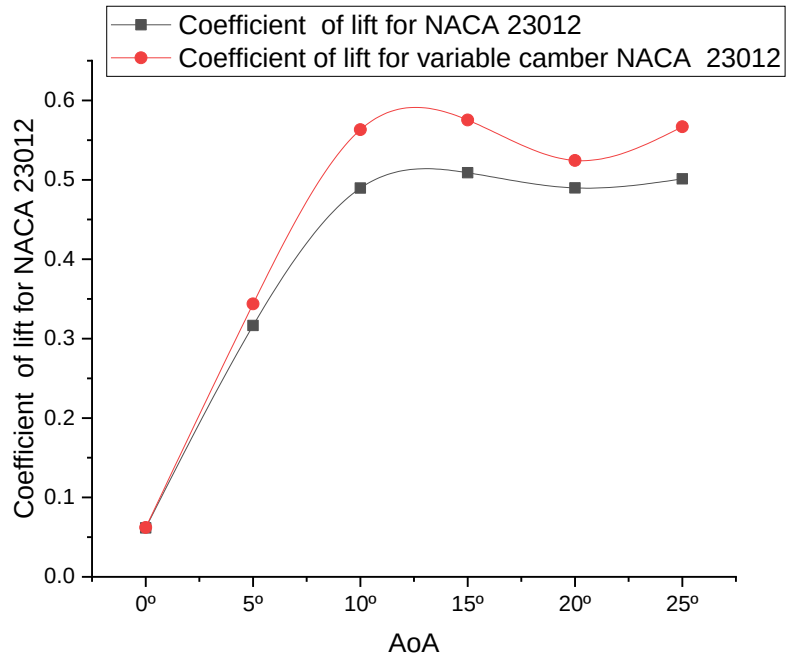


Fig 6-17 Coefficient of lift variation at 30 m/sec

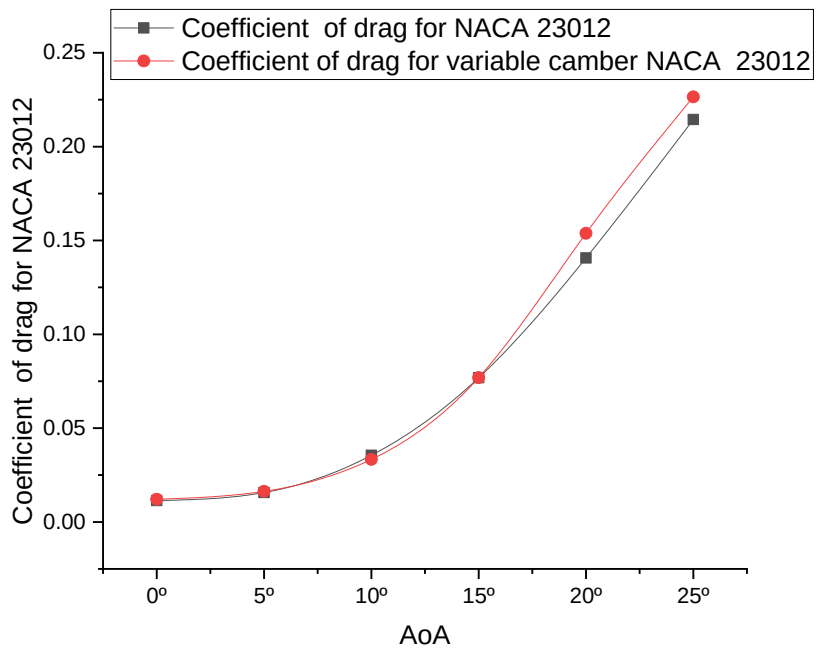


Fig 6-18 Coefficient of drag variation at 30 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 35 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.06141  | 0.316273 | 0.487297 | 0.504354 | 0.487058 | 0.499408 |
| <b>CL Vc 23012</b> | 0.061972 | 0.343522 | 0.561739 | 0.571275 | 0.521715 | 0.565605 |
| <b>CD 23012</b>    | 0.011329 | 0.015579 | 0.035229 | 0.077564 | 0.140666 | 0.2146   |
| <b>CD Vc 23012</b> | 0.012071 | 0.016148 | 0.032931 | 0.076866 | 0.153938 | 0.226884 |

Table 6-10  $C_l$  and  $C_d$  value for NACA 23012 and variable camber NACA 23012

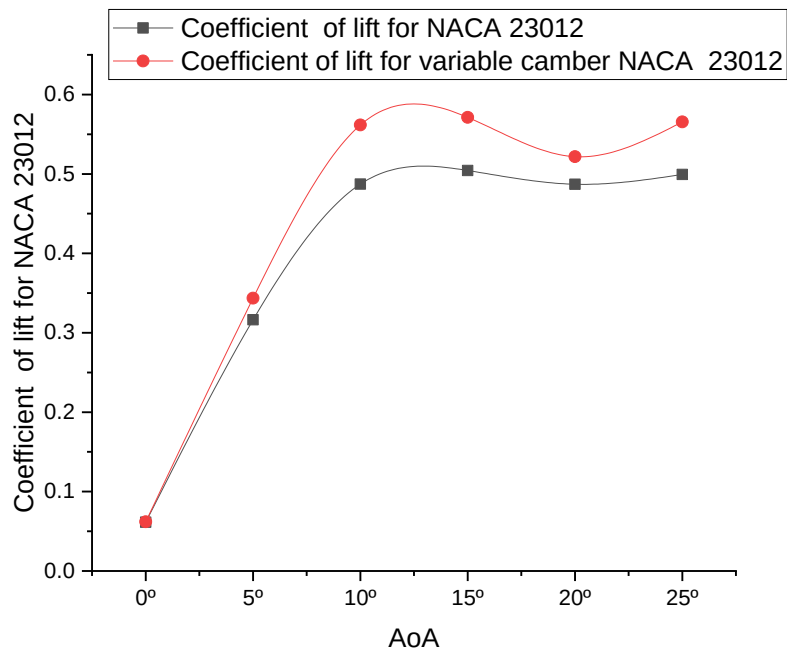


Fig 6-19 Coefficient of lift variation at 35 m/sec

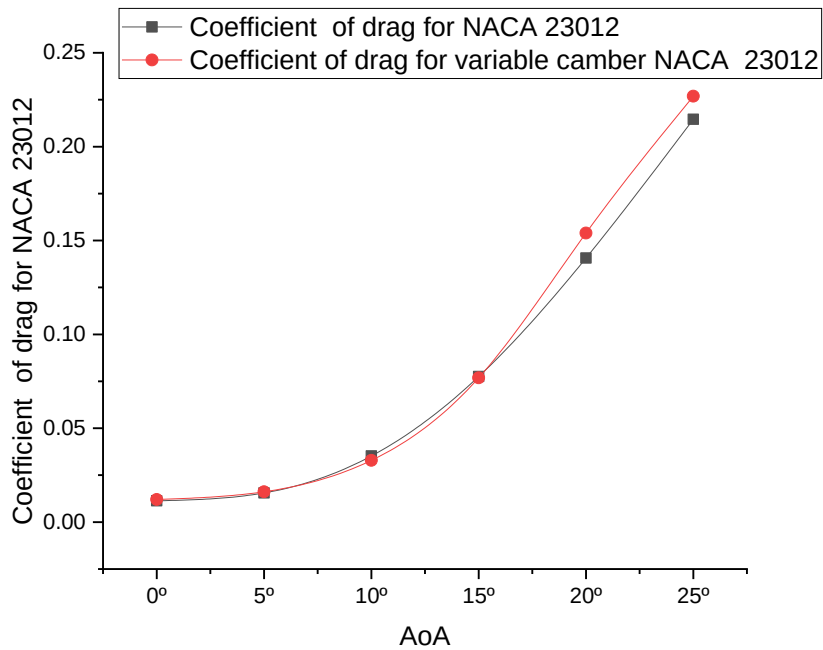


Fig 6-20 Coefficient of drag variation at 35 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 40 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.061264 | 0.315965 | 0.485136 | 0.500076 | 0.48491  | 0.497987 |
| <b>CL Vc 23012</b> | 0.061863 | 0.343122 | 0.560194 | 0.567212 | 0.519539 | 0.564665 |
| <b>CD 23012</b>    | 0.01132  | 0.015521 | 0.035049 | 0.078925 | 0.140689 | 0.214754 |
| <b>CD Vc 23012</b> | 0.012085 | 0.016112 | 0.032673 | 0.077445 | 0.154062 | 0.227014 |

Table 6-11  $C_l$  and  $C_d$  for NACA 23012 and variable camber NACA 23012

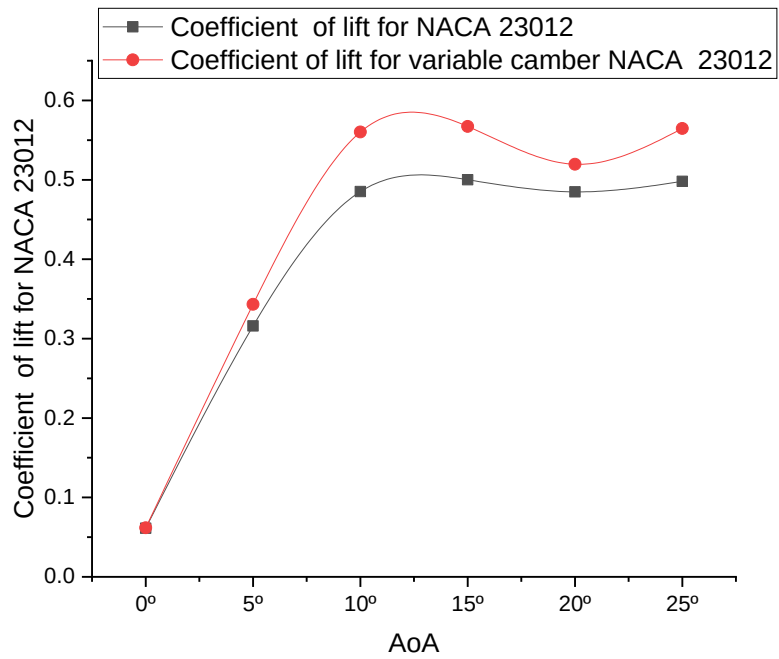


Fig 6-21 Coefficient of lift variation at 40 m/sec

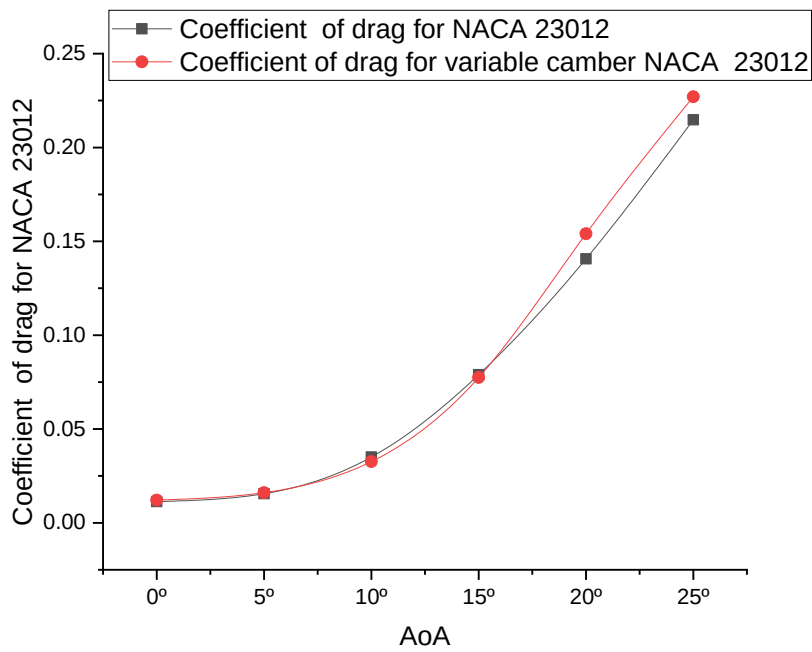


Fig 6-22 Coefficient of drag variation at 40 m/sec

Plotting  $C_l$  and  $C_d$  at velocity 45 m/sec NACA 23012 and variable camber NACA 23012

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.0611   | 0.315525 | 0.482933 | 0.499347 | 0.483057 | 0.496763 |
| <b>CL Vc 23012</b> | 0.061758 | 0.342688 | 0.558582 | 0.563265 | 0.517688 | 0.563824 |
| <b>CD 23012</b>    | 0.011361 | 0.015525 | 0.034955 | 0.077782 | 0.140739 | 0.214893 |
| <b>CD Vc 23012</b> | 0.01215  | 0.016139 | 0.032521 | 0.077469 | 0.154177 | 0.227158 |

Table 6-12  $C_l$  and  $C_d$  for NACA 23012 and variable camber NACA 23012

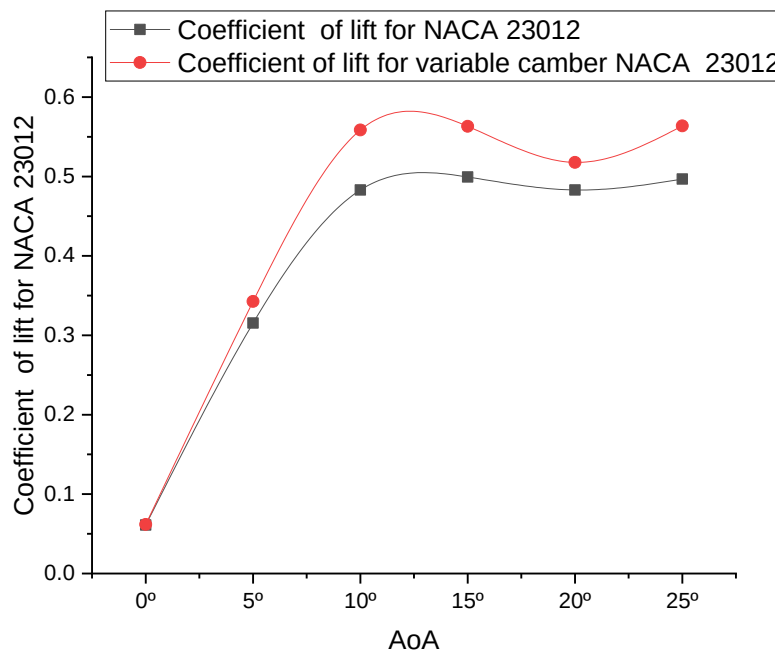


Fig 6-23 Coefficient of lift variation at 45 m/sec

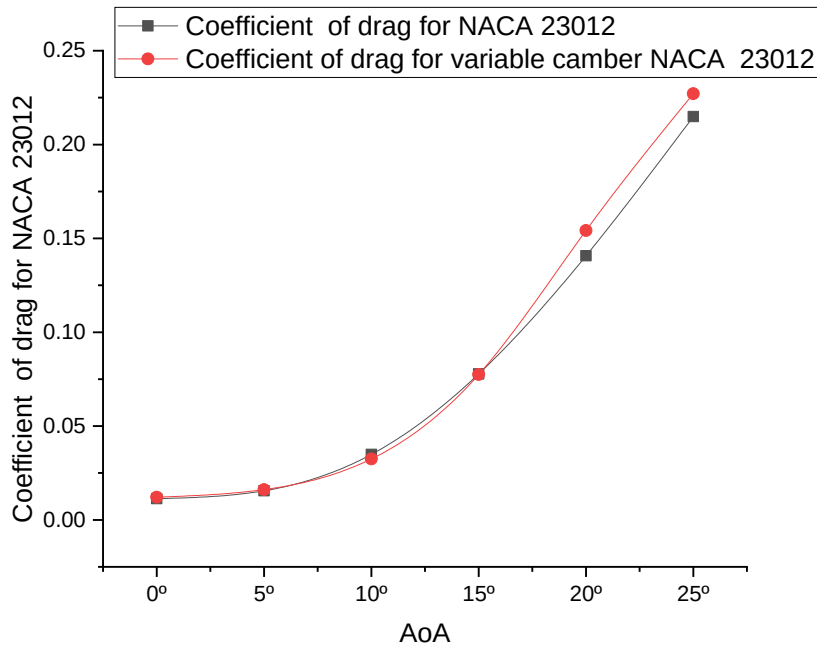


Fig 6-24 Coefficient of drag variation at 45 m/sec

### **6.1.3 Increment in Lift and Drag with variable camber mechanism**

It is important to calculate the Lift and the value of drag with empirical formula so that we can analyze the increment in lift and drag while using a variable camber mechanism; as we know from wind tunnel and Ansys simulation, there is an increment in coefficient of lift and drag values at respective AoA and velocities. Lift and Drag are directly proportional to density, the square of the speed, the surface area of the wing

#### **An empirical formula for lift and drag**

$$\text{Lift} = C_l \times \rho \times V^2 \times S$$

$$\text{Drag} = C_d \times \rho \times V^2 \times S$$

Where  $\rho$  is density and  $V$  is the velocity at which the experiment is conducted, and  $S$  is the plan surface area of the airfoil; here, for the experiment span of the airfoil is 45cm and the chord

length is 19cm and while deploying variable camber mechanism the chord length changes to 21 cm respectively during the introduction of variable chamber mechanism.

Again during the analysis, we have conducted experiments at the various angle of attack and velocities; we are more interested in percentage increment in Lift and drag during the highest AoA and velocity because we will be deploying a variable camber mechanism at take-off and landing, which usually happens higher AoA and velocities. So here, we get the following results for calculating increment in lift and drag considering an AoA of 20° and 10°, which is the highest for any aircraft and at a velocity of 45 m/sec.

|                    | 0° AoA   | 5° AoA   | 10° AoA  | 15° AoA  | 20° AoA  | 25° AoA  |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.0611   | 0.315525 | 0.482933 | 0.499347 | 0.483057 | 0.496763 |
| <b>CL Vc 23012</b> | 0.061758 | 0.342688 | 0.558582 | 0.563265 | 0.517688 | 0.563824 |

Plotting Cl at velocity 45 m/sec NACA 23012 and variable camber NACA23012

#### **Percentage Increment of Lift while deploying variable camber mechanism**

$$\begin{aligned}\% \text{ Lift increment at } 20^\circ \text{ for VC NACA 23012} &= \frac{606779.67 - 512266.098}{606779.67} \times 100 \\ &= 15 \%\end{aligned}$$

$$\begin{aligned}\% \text{ Lift increment at } 10^\circ \text{ for VC NACA 23012} &= \frac{654711.34 - 512134.60}{654711.34} \times 100 \\ &= 21 \%\end{aligned}$$

Similarly, for increment in drag only considering drag at 25°, which is the maximum angle of attack at which analysis is done considering the data mentioned in the table below, we get

| Angle of attack    | 0°       | 5°       | 10°      | 15°      | 20°      | 25°      |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CD 23012</b>    | 0.011361 | 0.015525 | 0.034955 | 0.077782 | 0.140739 | 0.214893 |
| <b>CD Vc 23012</b> | 0.01215  | 0.016139 | 0.032521 | 0.077469 | 0.154177 | 0.227158 |

Plotting Cd at velocity 45 m/sec NACA 23012 and variable camber NACA 23012

### Percentage increment in Drag while deploying variable camber mechanism

$$\begin{aligned} \text{\% of Increment in drag at } 25^\circ \text{ for VC NACA 23012} &= \frac{266250.82 - 227886.97}{266250.82} \times 100 \\ &= 14 \% \end{aligned}$$

### 6.2 Error analysis

Since we have experimental and computational data available, an error analysis is done by comparing both data at a particular velocity and AoA; since we have carried out experiments at different velocities and various angles of attack, for conducting error analysis here, we will only consider the value of the coefficient of lift at 10° at 45 m/sec since this the maximum velocity at which test is conducted. The maximum error probability is possible at the highest flow velocity, so considering data at a flow velocity of 45 m/sec.

| Simulation data                                                            |
|----------------------------------------------------------------------------|
| Plotting Cl at velocity 45 m/sec NACA 23012 and variable camber NACA 23012 |

|                    | 0° AoA   | 5° AoA   | 10° AoA  | 15° AoA  | 20° AoA  | 25° AoA  |
|--------------------|----------|----------|----------|----------|----------|----------|
| <b>CL 23012</b>    | 0.0611   | 0.315525 | 0.482933 | 0.499347 | 0.483057 | 0.496763 |
| <b>CL Vc 23012</b> | 0.061758 | 0.342688 | 0.558582 | 0.563265 | 0.517688 | 0.563824 |

| <b>Wind tunnel data</b>                                                    |  |
|----------------------------------------------------------------------------|--|
| Plotting Cl at velocity 45 m/sec NACA 23012 and variable camber NACA 23012 |  |

|                    | <b>0° AoA</b> | <b>5° AoA</b> | <b>10° AoA</b> | <b>15° AoA</b> | <b>20° AoA</b> | <b>25° AoA</b> |
|--------------------|---------------|---------------|----------------|----------------|----------------|----------------|
| <b>CL 23012</b>    | 0.063544      | 0.328146      | 0.50225        | 0.519321       | 0.502379       | 0.516634       |
| <b>CL Vc 23012</b> | 0.064228      | 0.356396      | 0.580925       | 0.585796       | 0.538396       | 0.586277       |

### **Calculation of percentage of Error for variable camber NACA 23012**

$$\begin{aligned}
 \% \text{ of Error} &= \frac{\text{Variable camber data at wind tunnel} - \text{Variable camber data at Simulation}}{\text{Variable camber data at the wind tunnel}} \times 100 \\
 &= \frac{0.580925 - 0.558582}{0.580925} \times 100 \\
 &= 3.84 \%
 \end{aligned}$$

### **Calculation of percentage error for NACA 23012**

$$\begin{aligned}
 \% \text{ of Error} &= \frac{\text{NACA 23012 data at wind tunnel} - \text{NACA 23012 data at Simulation}}{\text{Variable camber data at the wind tunnel}} \times 100 \\
 &= \frac{0.50225 - 0.482993}{0.50225} \times 100 \\
 &= 3.98 \%
 \end{aligned}$$

### **6.3 Conclusion**

Morphing is a major method that helps aircraft make changes to their components, such as their wings, to adapt or adjust to their varying flight conditions for efficient control and performance. In this paper, a mechanism is designed that aims at using morphing techniques to mimic a bird's tail. The main purpose of this design is to achieve yaw control and directional stability in the absence of a rudder and with the aid of the horizontal stabilizer solely. Our next step was to design a suitable mechanism to vary the chord length and then manufacture the airfoils with variable camber mechanism testing them in the wind tunnel and Ansys simulation software at different low subsonic speeds and calculating the lift achieved by using the empirical lift formulae and comparing the results observed computationally and experimentally received. This work highlights the different types of in-plane and out plane morphing techniques developed in the past decades, mechanisms used, material, and the development and incorporation of these techniques in modern-day aircraft. It extends the research on airfoils with varying the camber along the chord line. Our first challenge was to narrow it down to a particular NACA series since there is a lot of NACA series available; our review of the articles and research led us to NACA 5 digit series, and further, we narrowed it down to NACA 23012 series following observations were noted down for NACA 23012 series. Extending the research on NACA 23012 airfoil using a variable camber mechanism to increase camber, we find that just increasing the chord length by a mere 2 cm will enhance lift to 21% at  $10^\circ$  which is quite promising. The corresponding drag increment was just 14% at  $25^\circ$  angle of attack. We also find that there is an error when we compare simulation data with experimental data on airfoils mainly due to the difference in an ideal atmospheric condition which is quite possible in software to the real atmospheric condition in the wind tunnel. We found an error of 4 % (approx) while comparing simulation and experimental data at a given angle of attack. In future, we will see a lot of new morphing concepts which will be capable of changing the wing design, and we may choose or possibly imitate nature; with the world gradually changing from gas turbine propulsion to electric propulsion, it's quite possible that large portion of the wings which is now occupied for fuel tanks will be available for morphing and variable camber mechanism and possibly new emerging techniques will take over in the near future as if now variable camber mechanism and morphing concept on the wings is a promising area to continue research.

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### **List of Publications**

|   |                                                                                                                                   |          |                                     |
|---|-----------------------------------------------------------------------------------------------------------------------------------|----------|-------------------------------------|
| 1 | A review on Aerodynamic performance of NACA Airfoil for various Reynolds number                                                   | Review   | DOI10.1088/17426596/1473/1/012003   |
| 2 | Computational study on chamber morphing wing concept for efficient lift at various angle of attack                                | Research | DOI: 10.1109/ICPS51508.2020.00020   |
| 3 | Aerodynamics and flow separation investigation of NACA2412 and NACA23012 airfoils with dual vents at predefined separation region | Research | DOI: 10.1063/5.0035079              |
| 4 | Albatross and Falcon inspired Bionic UAV: An Aerodynamic Analysis                                                                 | Research | DOI:10.1.15394/ijaaa.2021.1598      |
| 5 | Evolutionary and Hereditary Traits of an Albatross and its Aerodynamic Optimality                                                 | Research | DOI: 10.1088/1755-1315/775/1/012002 |