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Numerical Analysis and Estimation of Aerodynamic Performance of a Geometric Twisted Wing Configuration using NACA 23012, NACA 23015 and NACA 23018

Pritam Saha

Student, School of Mechanical Engineering, Department of Aerospace Engineering, Lovely Professional University, Jalandhar, India

Abstract: Airfoil is the cross-section shape of wing which generates a significant magnitude of lift force in an airflow, it is the most important part of an aircraft. In this study a geometrical twisted wing configuration is designed using different airfoils in which NACA 23012 is used at tip of the wing, NACA 23015 is used at mid span and NACA 23018 is used at the root of the wing. ANSYS is used for the creation of geometry and meshing and FLUENT is used as a solver to numerically analysis the airfoils of wing configuration. The aerodynamic performance parameter of the wing is estimated using the turbulence model.

Index Terms: Airfoil, NACA 23012, NACA 23015, NACA 23018, Geometric twisted wing, ANSYS Fluent

I. INTRODUCTION

Wing is a mechanical device having a streamlined cross section which produce lift due to the aerodynamic forces acting on it. Lift produce by wing depends on the airfoil shape, angle of attack and free stream velocity. Wing consist of different control surface and high lift device.

Airplanes can be classified on the basis of wing configuration called as high wing, mid wing, low wing and on the basis of wing shape it can be classified as straight wing, sweep wing etc.

The cross-section of a typical airplane wing is an airfoil and is primarily responsible for producing the lift force which balances the weight of the aircraft during flight. These airfoil shapes are described by a series of digits following the word "NACA". The parameters in the numerical code can be entered into equations to precisely generate the cross-section of the airfoil and calculate its properties.

First digit of a 5-digit NACA expressed the lift coefficient multiplied by 3/20. Second digit express the maximum chamber position. Third digit express that the chamber line is normal or reflex and last digit express the maximum thickness as percentage of the chord.

In this study a geometric twisted wing configuration is designed using different airfoil and its performance is estimated using computational fluid dynamics (CFD) at different conditions. NACA 23018 is used at the root of the wing, NACA 23015 is used at the mid span and NACA 23012 is used at the tip of the wing structure. Using ANSYS FLUENT these airfoils aerodynamic performance is obtained and the interaction of wing airfoil and fluid flow is plotted.

II. COMPUTATIONAL METHOD

In this paper NACA 23012, NACA 23015 and NACA 23018 airfoils are utilized. All the airfoils are cambered airfoil and having different thickness as 12% thickness of the chord for NACA 23012, 15% thickness of the chord for NACA 23015 and 18% thickness of the chord for NACA 23018.

Aerodynamic performance parameters are obtained for airfoils at different angle of attack where Reynolds number is 3×10^6 for these simulation.

Density of the air at given temperature is $\rho = 1.225 \text{ kg/m}^3$ and the viscosity is $\mu = 1.7894 \times 10^{-5} \text{ kg/(m * s)}$. The turbulent viscosity is computed through K-omega SST model.

III. GEOMETRY AND DOMAIN GENERATION

Geometry of these airfoils are designed in ANSYS Geometry where the Chord of the airfoil is 1 meter. The computational domain or wind tunnel setup is extended 15m upstream of the leading edge of the airfoil, 15m downstream of the trailing edge, and 20m above the pressure surface. The velocity boundary condition at the inlet of the domain is 43.822 m/s (u). The x-component of velocity is calculated by $x = u \times \cos(\alpha)$ and the y component of velocity is calculated by $y = y \times \sin(\alpha)$, where α is the angle of attack in degrees. In this study it is assumed that inlet velocity is less turbulent than pressure outlet. Hence, for velocity inlet boundary condition turbulence intensity is considered 0.1%.

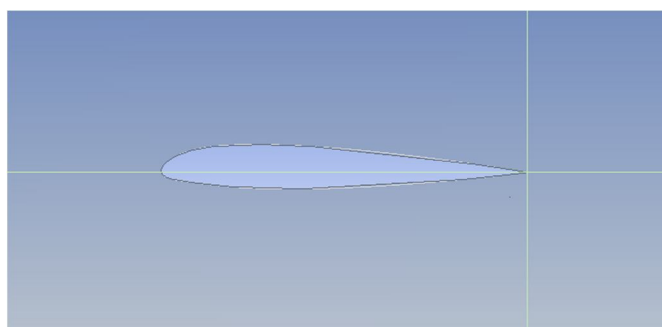


Figure 1 : Geometry of NACA 23012

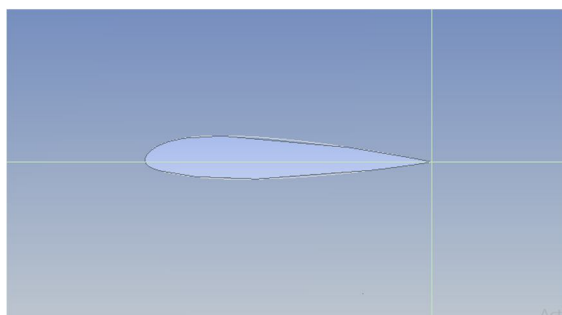


Figure 2: Geometry of NACA 23015

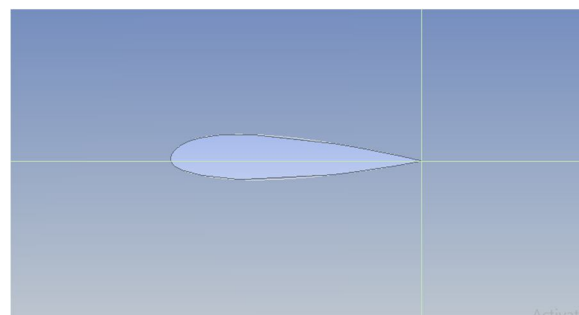


Figure 3: Geometry of NACA 23018

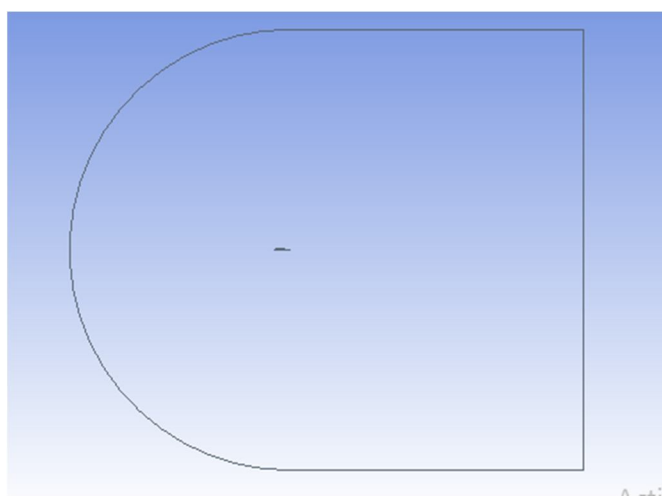


Figure 4: Geometry of Domain

IV. MESHING GENERATION

Mesh is a process to divide a component into very small elements or divide into nodes. ANSYS MESH is used to mesh the domain and split into 4 parts for better mesh quality. The model is shown in figure below for both airfoil and domain mesh.

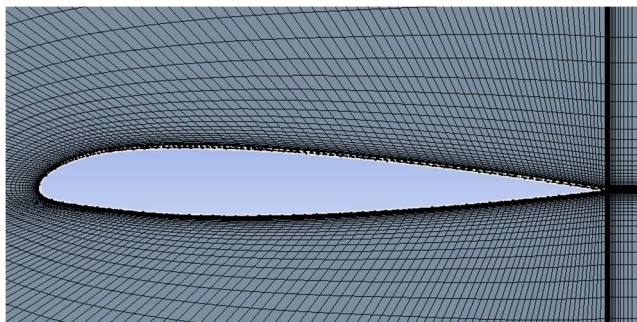


Fig 5: Mesh quality around NACA 23012

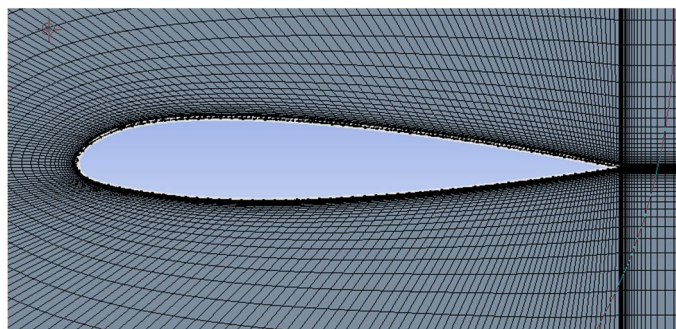


Fig 6: Mesh quality around NACA 23015

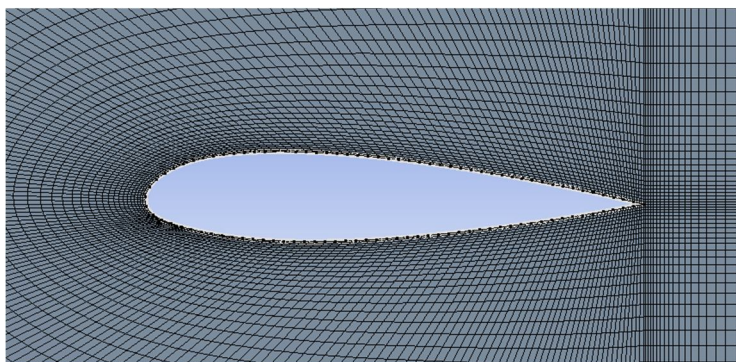


Fig 7: Mesh quality around NACA 23018

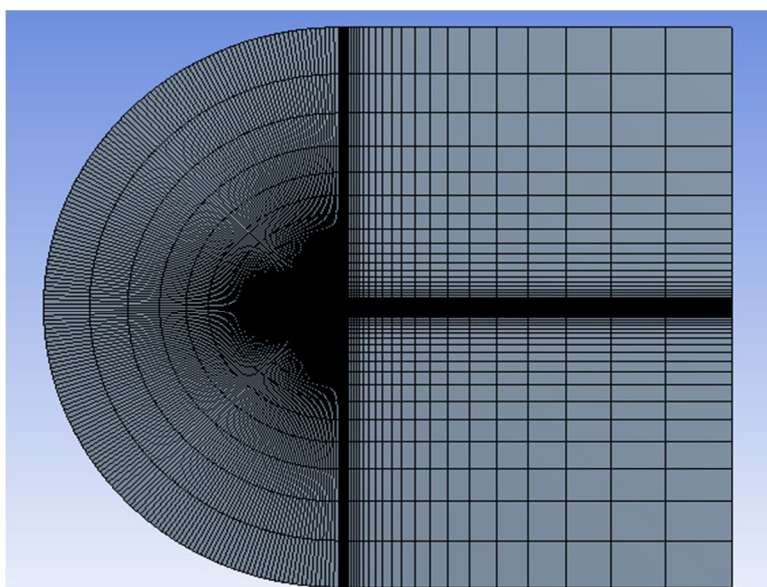


Fig 8: Mesh of the Domain

V. COMPUTATIONAL PROCESS ON FLUENT

Turbulence model is used to predict the aerodynamic parameters of the wing at different airfoil section using ANSYS FLUENT where double precision is used and K- ω SST turbulence model is used in this study to predict the Aerodynamic characteristics on all the airfoils.

K- ω model is a two-equation turbulence model which is used for RANS equations. It uses two variables, k which is the kinetic energy and ω which is the specific rate of dissipation. This model is combined with SST model which is Shear Stress Transport which is also widely used. The input parameters along with specific conditions as indicated in the table below were given on FLUENT.

Solver	Pressure based steady
Viscous Model	K- ω SST model
Density(kg/m ³)	1.225
Viscosity(kg/m-s)	1.7894e-05
Turbulence intensity ratio	0.1
Turbulence length scale	0.3
Inlet Velocity(m/s)	43.822
Reynolds Number	3 x 10 ⁶
Chord length(m)	1
Momentum	Second Order Upwind
Pressure velocity coupling	Coupled

Table 1: Fluent details

VI. RESULT OF NEUMERICAL ANALYSIS

A. Pressure Contour

In this study from the computational analysis pressure contour is obtained at different angle attack are given below

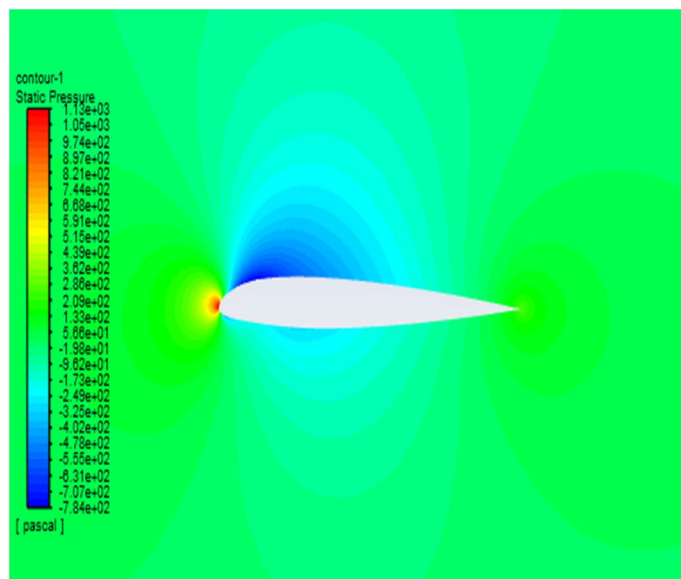


Fig 9: Pressure contour for NACA 23012 at 0° angle of attack

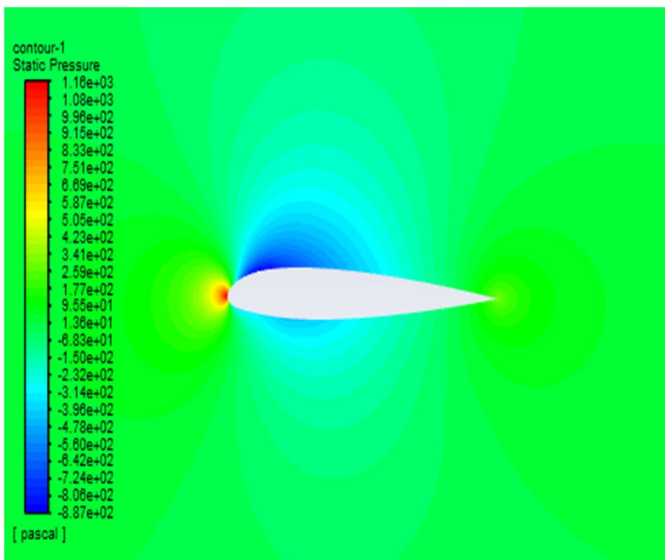


Fig 10: Pressure contour for NACA 23015 at 0° angle of attack

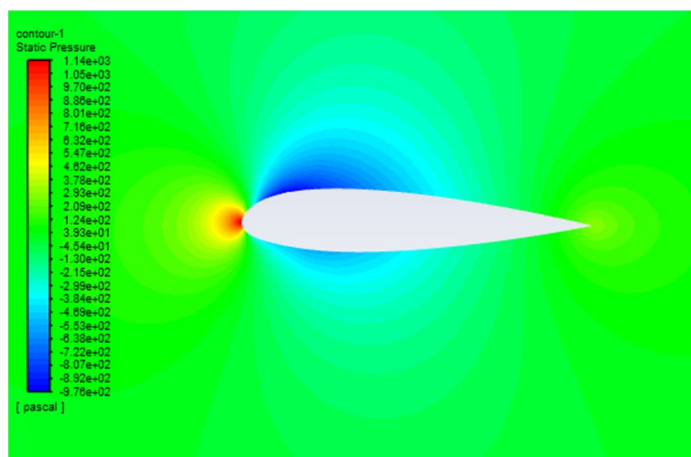


Fig 11: Pressure contour for NACA 23018 at 0° angle of attack

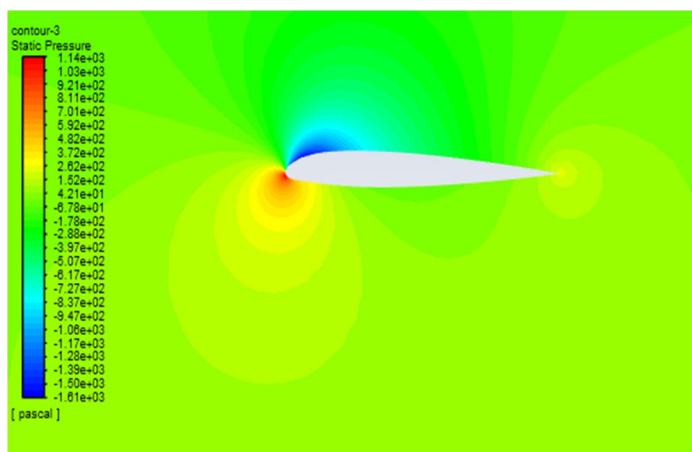


Fig 12: Pressure contour for NACA 23012 at 4° angle of attack

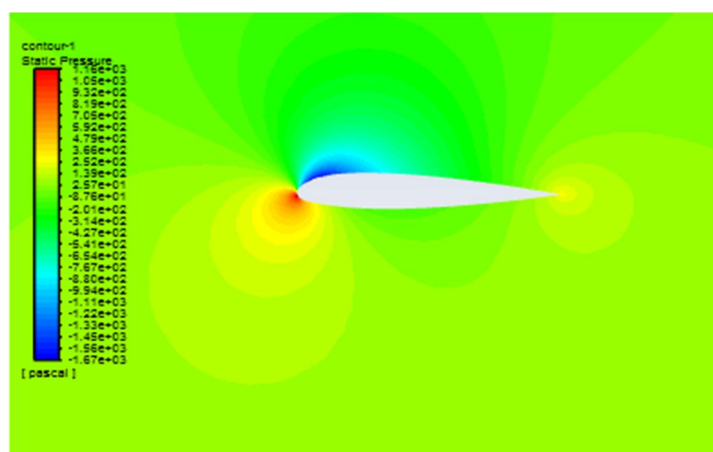


Fig 13: Pressure contour for NACA 23015 at 4° angle of attack

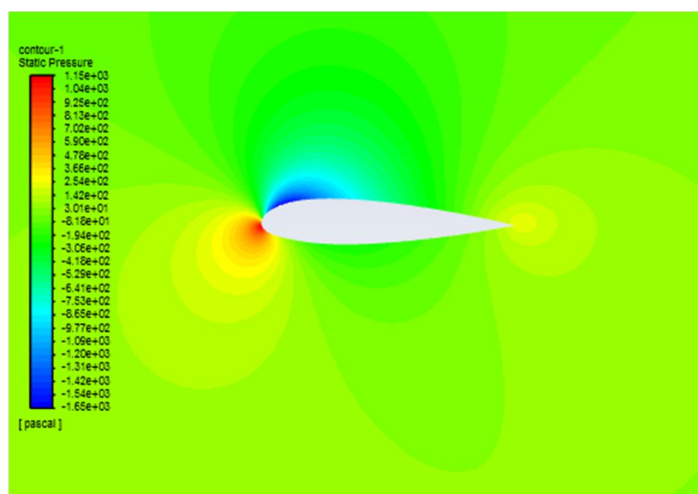


Fig 14: Pressure contour for NACA 23018 at 4° angle of attack

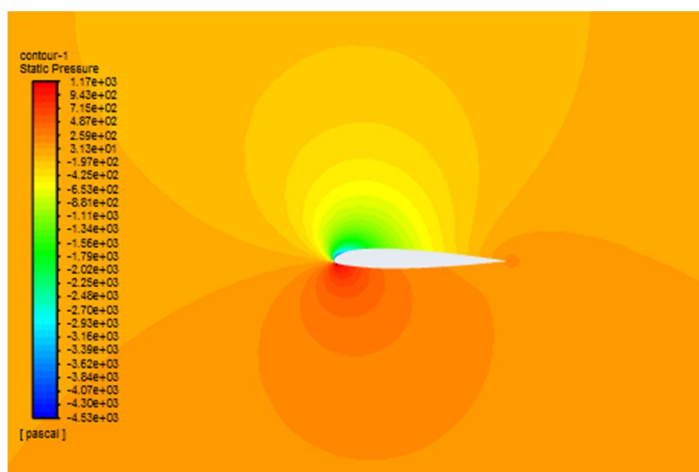


Fig 15: Pressure contour for NACA 23012 at 10° angle of attack

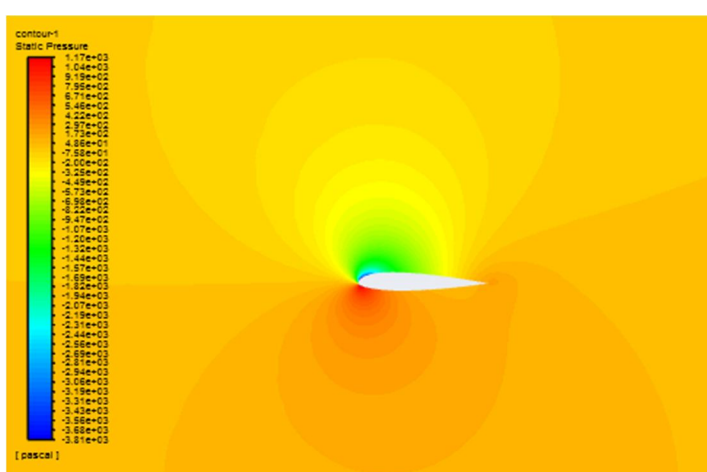


Fig 16: Pressure contour for NACA 23015 at 10° angle of attack

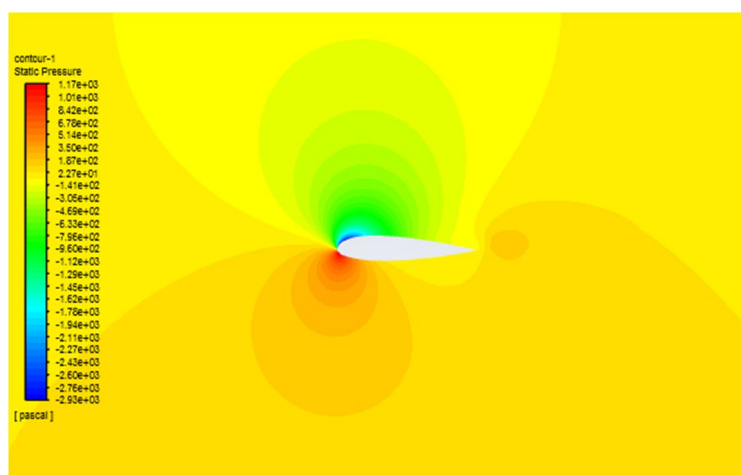


Fig 17: Pressure contour for NACA 23018 at 10° angle of attack

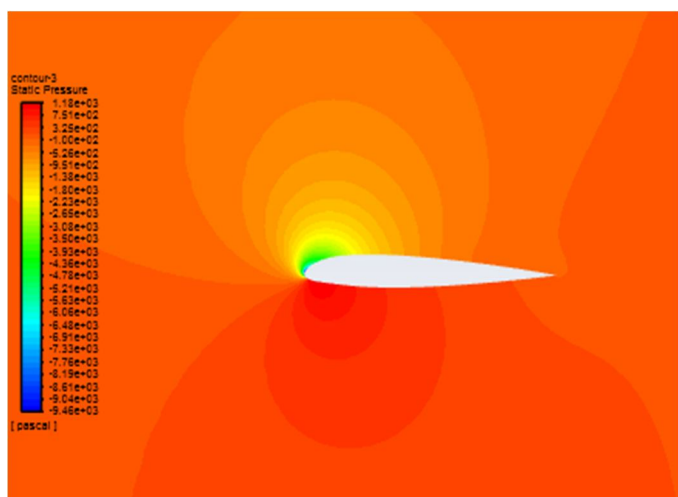


Fig 18: Pressure contour for NACA 23012 at 16° angle of attack

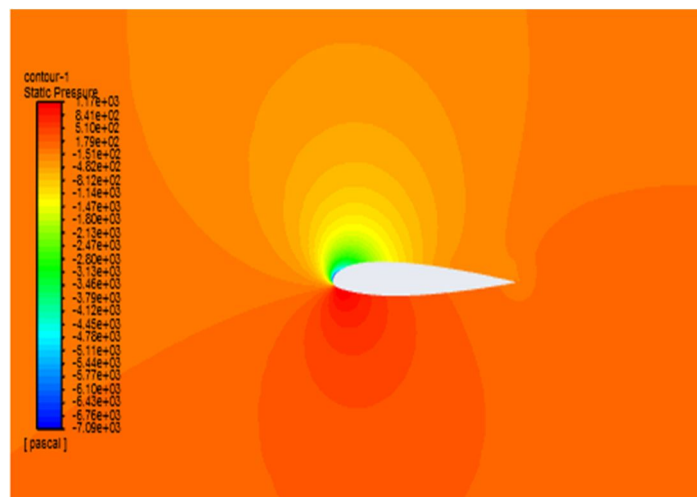


Fig 19: Pressure contour for NACA 23015 at 16° angle of attack

B. Velocity Contour

In this study from the computational analysis pressure contour is obtained at different angle attack are given below

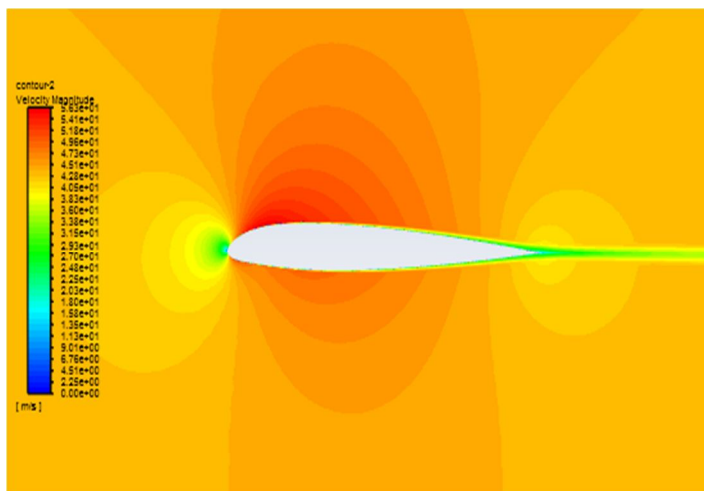


Fig 20: 1Velocity contour for NACA 23012 at 0° angle of attack

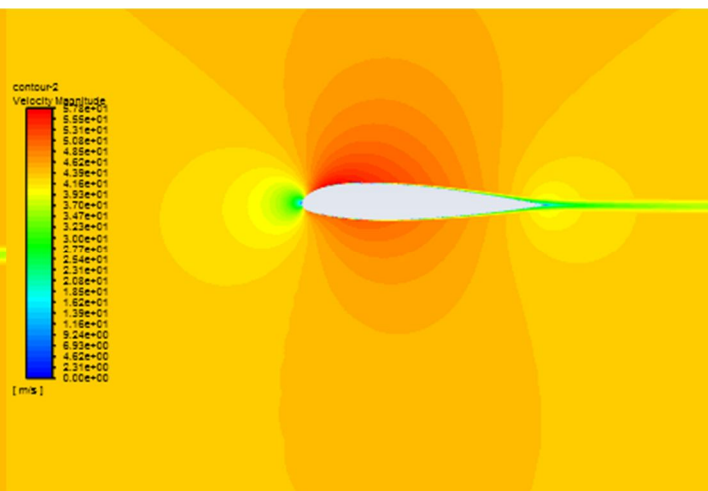


Fig 21: Velocity contour for NACA 23015 at 0° angle of attack

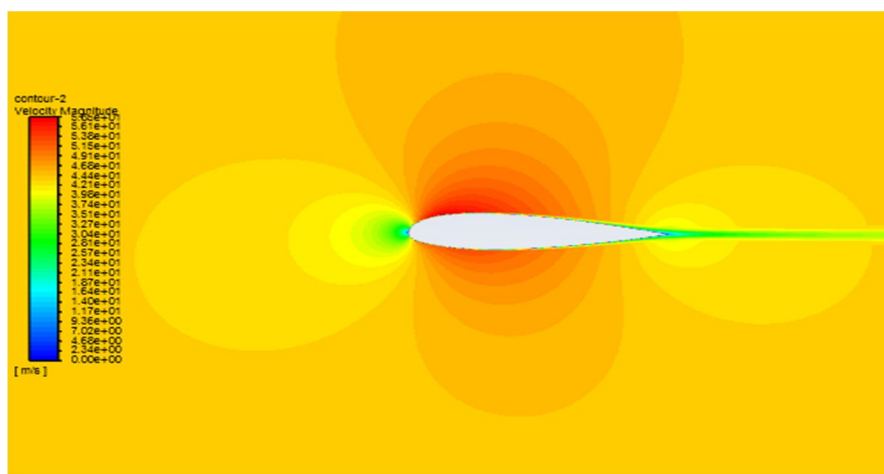


Fig 22: Velocity contour for NACA 23018 at 0° angle of attack

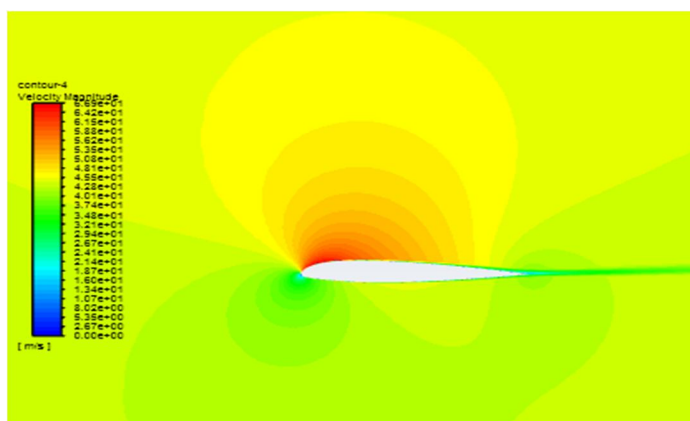


Fig 23: Velocity contour for NACA 23012 at 4° angle of attack

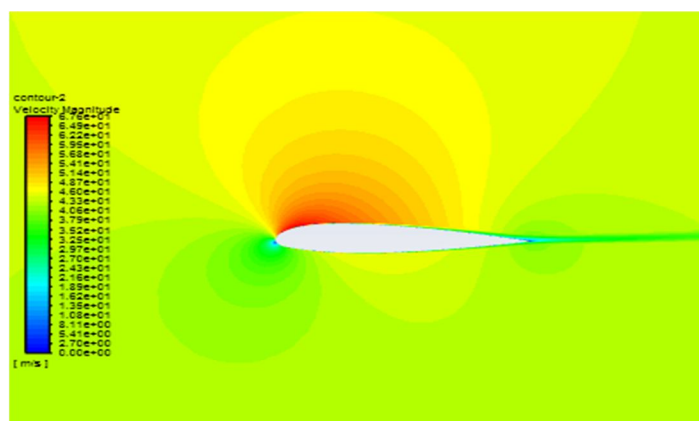


Fig 24: Velocity contour for NACA 23015 at 4° angle of attack

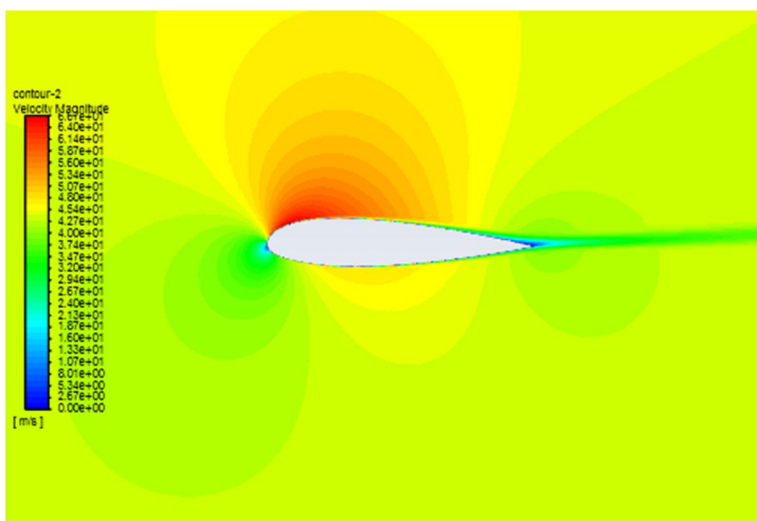


Fig 25: Velocity contour for NACA 23018 at 4° angle of attack

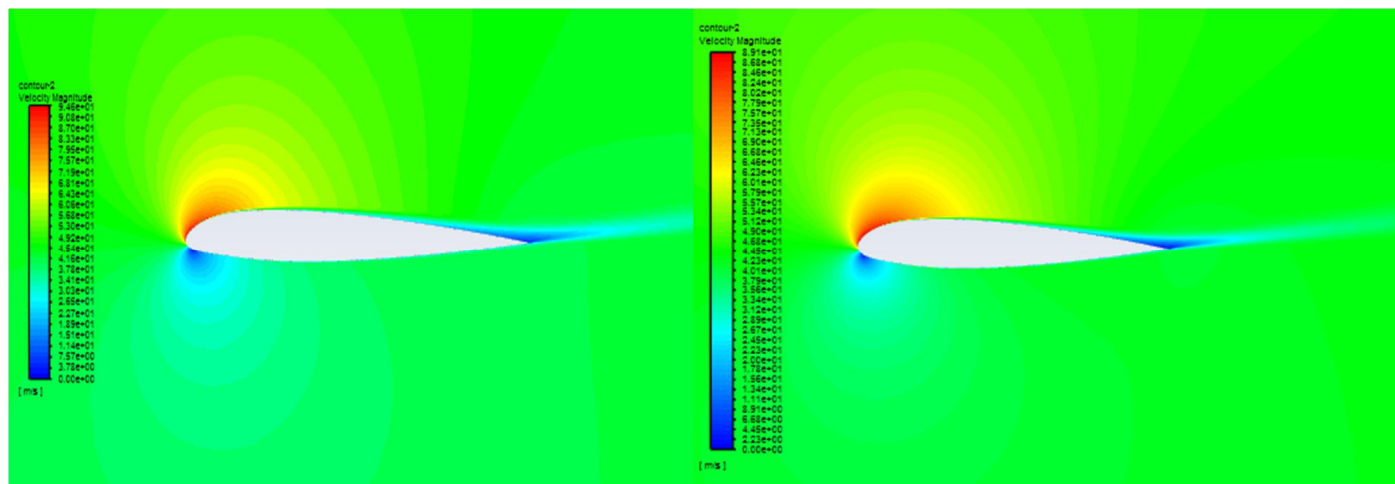


Fig 26: Velocity contour for NACA 23012 at 10° angle of attack

Fig 27: 2Velocity contour for NACA 23015 at 10° angle of attack

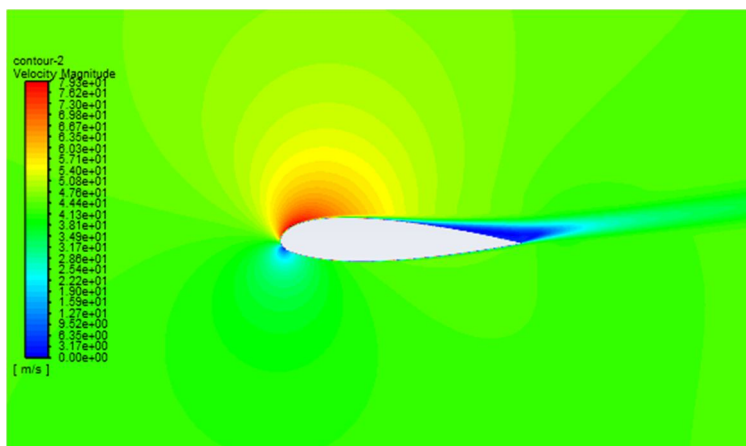


Fig 28: Velocity contour for NACA 23018 at 10° angle of attack

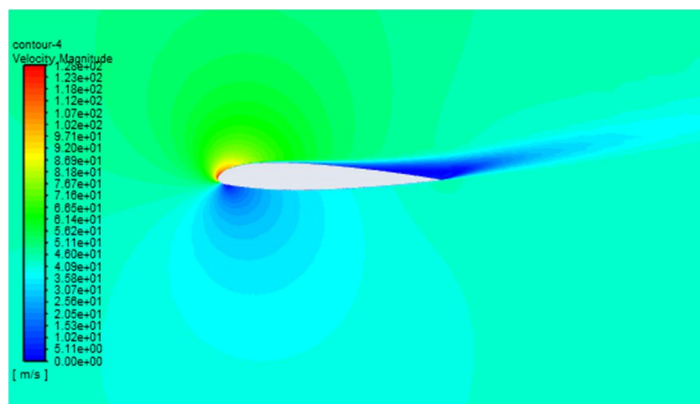


Fig 29: Velocity contour for NACA 23012 at 16° angle of attack

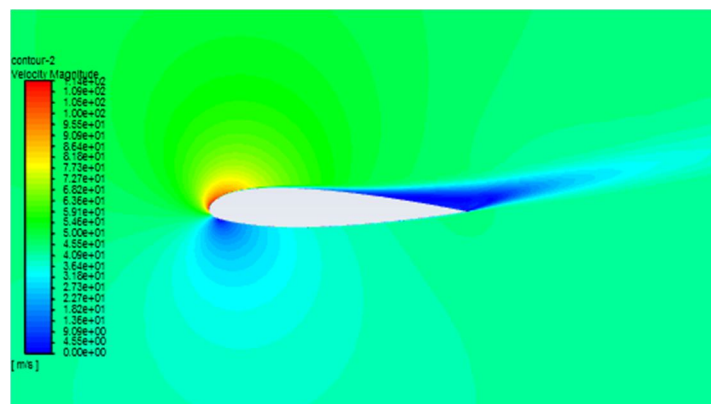


Fig 30: Velocity contour for NACA 23015 at 16° angle of attack

C. Comparison Plot of Coefficient of Lift

In this study the coefficient of lift and coefficient of drag at different angle of attack is estimated using computational approach and Coefficient of Lift Vs Angle of Attack and Coefficient of Drag Vs Angle of Attack is plotted below. From the this study it can be observed that the Coefficient of lift increases with the increase of angle of attack and at a point it certainly decreases this point is known as Stall angle of airfoil and at this point the maximum coefficient can be obtained.

Table 2: Coefficient of lift of NACA 23012, NACA 23015 and NACA 23018 at different angle of attack

Angle of Attack (°)	Coefficient of Lift NACA 23012	Coefficient of Lift NACA 23015	Coefficient of Lift NACA 23018
-1	0.017903282	0.018948338	0.025917047
0	0.1255222	0.12450898	0.11564582
2	0.34069638	0.33522487	0.29543425
4	0.59375893	0.5439986	0.47798804
6	0.76311953	0.74833825	0.65412238
8	0.96328324	0.94496006	0.80298832
10	1.1509004	1.1279163	0.87747043
12	1.3179977	1.2920731	0.71487916
14	1.4497315	1.4183313	0.67936626
16	1.4976219	1.4475068	0.65935424
18	0.97698109	0.93391762	0.61311952

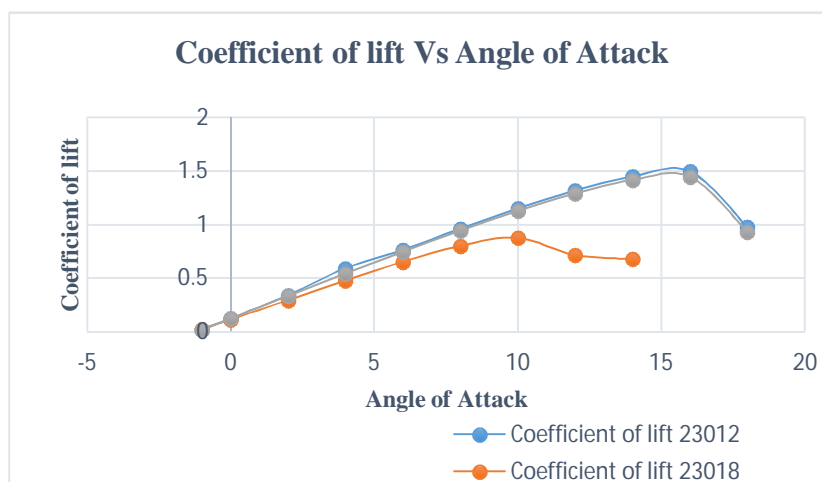


Figure 31: Coefficient of lift Vs Angle of attack

VII. ESTIMATION OF THE MAXIMUM COEFFICIENT OF THE LIFT

In this study of wing different airfoils are used at different position of the wing. In the geometrically twisted wing NACA 23012 airfoil is used at the tip of the wing, NACA 23015 is used at mid span and NACA 23018 is used at the root of the wing. In all the airfoils it can be observed that the maximum chamber is placed at 15% of the chord length and the thickness are 12 %, 15% and 18% respectively. Using of these different thickness of airfoil produce structural strength to the wing and aerodynamic stability during flight. It can be observed from the computational experimental data the NACA 23018 Stalls at 10° angle of attack at this point maximum coefficient of lift is 0.87 and NACA 23012 and NACA 23015 stalls at 14° angle of attack and maximum coefficient of lift is 1.49 and 1.44 respectively.

Taking the average from the airfoils maximum coefficient of lift is obtained as 1.266 and we designing a wing with flap at trailing edge. From the historical data from different aircraft design we consider a flap with 45° deflection increase the maximum coefficient of lift 0.9 so the average maximum coefficient of lift becomes $(1.266+0.9) = 2.166$.

VIII. CONCLUSION

- A. It can be observed in the study that the flow separation takes place at different angle of attack in different airfoil section. The pressure generated on the lower surface of NACA 23018 airfoil is lesser than the NACA 23012 and NACA 23015 airfoil. Thus the lift generated on NACA 23012 and NACA 23015 is more as there is greater pressure difference on the upper and lower surface than NACA 23018 airfoil.
- B. It can be observed in this study that the thickness of the airfoil and the aerodynamic performance are related. Due to that NACA 23018 stalls first at 10° angle of attack and NACA 23012 and NACA 23015 stalls later at 16° angle of attack. As the root of the wing stalls first but the tip of the wing stall at higher angle of attack which help the control system to keep the airplane in stable condition.
- C. Using different shape of airfoil at different position of the wing increase the aerodynamic performance than a wing using single airfoil shape. It also increases the strength of the wing.

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ABOUT THE AUTHOR



Mr. Pritam Saha
B.Tech (Aerospace Engineering) (Pursuing), Lovely Professional University

2016pritamsaha@gmail.com



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