Journal of Mechanical and Aeronautical Engineering Research
[Online ISSN: 2456 8368]
Volume 1 Issue 3; December 2017; PP – 25 to 28
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CFD analysis of flow characteristics of NACA0012 airfoil using SU2

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***Abstract*— This paper presents the CFD analysis of the flow over a NACA 0012 Airfoil at different Mach number and at different angle of attacks using SU2 open source CFD code. The Mach number examined were 0.1, 0.3, 0.5, while the angle of attack(AOA) ranged from 5 to 15 degrees in 5 degree increments. This work was initiated to create an awareness of su2 software through computational simulation. The characteristics of the naca0012 airfoil operating at different mach number and at different AOA is analyzed by predicting the lift and drag measurements. Low Reynolds number is selected for the analysis. The Euler solver of open source CFD code is chosen for laminar flow analysis over naca0012 airfoil. The simulations are predicting the lift to drag ratio of naca0012 airfoil for specific conditions accurately.**

***Keywords*— NACA 0012; angle of attack; SU2; CFD.**

I. Introduction

The computational fluid dynamics field is enlightened by the development of some modern computational software packages which helps in solving computational problems easily thereby saving computational time. The Aerospace Design Lab (ADL) of the Department of Aeronautics and Astronautics at Stanford University is engaged in improving the capability of SU2 released under an open-source license [1, 2].

SU2 is an open source software. PDE analysis is done through this software. The framework is suitable to arbitrary sets of governing equations for solving analysis and design problems. The suite core consists of solvers like Reynolds-averaged Navier-Stokes (RANS) solver, compressible and incompressible Euler solver, Navier-Stokes solver capable of simulating many problems in aerospace and mechanical engineering.

SU2 also provides adjoint method for shape optimization problems and adaptive mesh refinement for specific goal. This software run with configuration file. By setting the configuration file for the particular problem, SU2 is way simple and time saving software for any kind of simulation problems. [3] Euler solver scheme efficacy is found by computing 2d flow past a NACA0012 airfoil under subsonic condition and the implicit formulation shows good convergence acceleration over other explicit procedures.

Douvi et al. [4] showed the aerodynamic behavior of airfoil at different angles of attack and low Reynolds number. In this paper the aerodynamic behavior of NACA0012 airfoil is analyzed using SU2 software. The airfoil is investigated for two different mach number and for various angle of attack. The lift and drag coefficients are evaluated from the computational results obtained. The static pressure contour over the airfoil is also used to study the behavior of the naca0012 airfoil.

II. Numerical Method

A. Grid generation

The grid used in mesh generation is orthogonal curvilinear mesh has been displayed in Fig. 1. It is used for the analysis of flow over the airfoil and suitable fine mesh is done to minimize the error which arise due to coarseness of the grid. Naca0012 airfoil is used for the generation of orthogonal curvilinear grid. The C-grid consist of 9604 cells and 9800 nodal points with two surface markers. One marker is airfoil contains 98 boundary elements and other one is the farfield marker containing 294 boundary elements.

B. SU2 solver

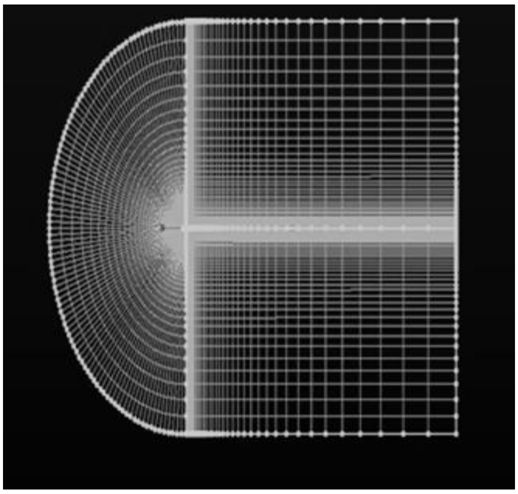

Fig.1 C-structured grid

The Stanford University Unstructured code is developed for Partial Differential Equation (PDE) analysis. SU2 CFD solver is designed primarily for solving computational fluid dynamics (CFD) and aerodynamic shape optimization problems. However, this code has a set of arbitrary equation to solve potential flow, hydrodynamics, electrodynamics, supersonic, hypersonic flow and chemically reacting flows. SU2 software is developed in Stanford university students providing an open-source license [1, 2].

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For this simulations, 5.0.0 “Raven” version of SU2 is used. Solver used is Euler solver. The Jameson-Schmidt-Turkel (JST) scheme and Euler implicit scheme are used to discretize the convective fluxes. Courant number is used to achieve the convergence easily thereby saving computational time. CFL number adopted for this analysis is 4. The weighted least square numerical method is used for spatial gradients. The non-dimensional coefficients are evaluated on the given surface markers.

III. Results and Discussions

To study the characteristics of airfoil using su2, NACA0012 airfoil is chosen for the analysis. Euler solver is chosen for computation. The airfoil characteristics like lift, drag is analyzed for different velocities and different angle of attack. Contours like pressure, pressure coefficient and mach number are plotted and the behavior of flow over the airfoil at different mach numbers is identified. The characteristics of the naca0012 airfoil is discussed below.

A. *Lift Characteristics*

Lift is predicted from the non-dimensional coefficients over the surface of the airfoil at angle of attack between 0^0 to 25^0. The airfoil is evaluated for mach number 0.1 and 0.5 and the lift coefficients are plotted for angle of attacks at 5-degree increment.

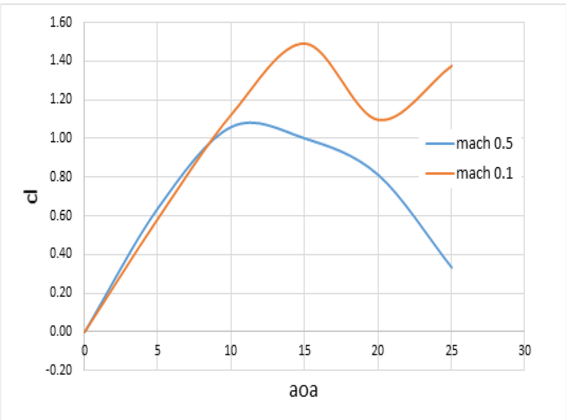

Fig.2 lift coefficient vs aoa

Fig.2 displays the lift coefficient of the airfoil at mach number 0.1 and 0.5. However, by changing the angle of attack the lift coefficient increases. For mach number 0.1 the cl increases from 0^0 to 15^0. While increasing the angle of attack beyond 16^0 stalling occurs and the cl value is decreased up to 20^0 angle of attack. Then while increasing the angle of attack furthermore the lift coefficient value increases. For mach number 0.5 the lift coefficient value increases up to 12^0 after that stalling occurs. Due to this stalling effect the lift coefficient decreases drastically even when the angle of attack is increased. Therefore, for low speed airfoil the lift coefficient in more when the mach number is low but when the mach number peaks up the lift coefficient decreases.

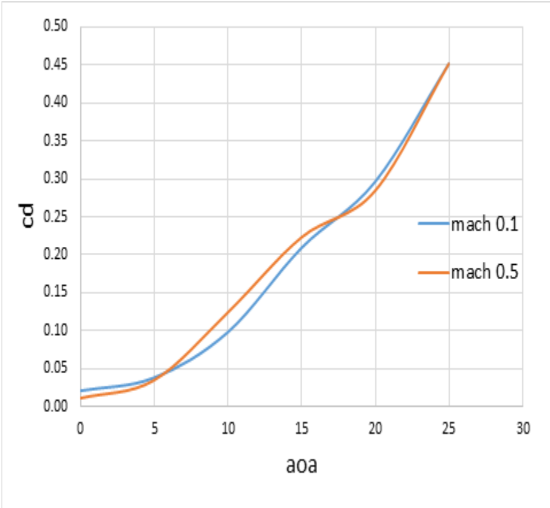

Fig.3 drag coefficient vs aoa

B. *Drag characteristics*

Fig.3 shows the drag coefficient of naca0012 airfoil at different angle of attack. The angle of attack is changed from 0^0 to 25^0 and simulation is done to analyze the behavior of airfoil at two mach number 0.1 and 0.5. For both mach numbers, the drag coefficient increases when the angle of attack increases. But for mach number 0.1 the drag coefficient value is less when the angle of attack is between 8^0 to 16^0 compared with the drag coefficient value at mach number 0.5.

C. *Pressure Contours*

The pressure over the surface of the naca0012 airfoil is displayed in the fig 4.a and 4.b. Pressure contours are evaluated for mach number 0.1 and 0.5 and for angle of attack between 0 to 25 degree. The flow stops at the leading edge portion of the airfoil which is called stagnation point where the pressure is very high. The red color region in the contour represent high pressure and the blue color region in the contour represent low pressure. For mach number 0.1, vortices formed adjacent to the trailing edge of the airfoil but it is minimum when the angle of attack is 0 degree. By changing the angle of attack the vortices are increased. For mach number 0.5, when the angle of attack is increased the vorticity diminishes.

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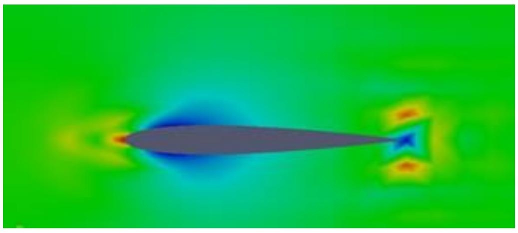
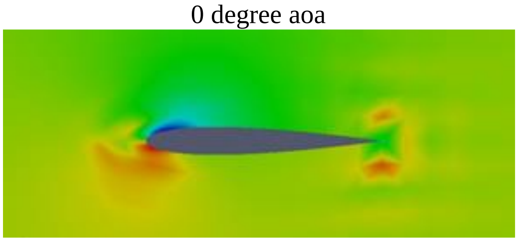
IV. CONCLUSION

The computational study of aerodynamic characteristics of naca0012 airfoil has been carried out using SU2 with Euler solver. The non-dimensional coefficients are found at angle of attack from 0 to 20 degree and at mach number 0.1 and 0.5. At mach number 0.1, the lift coefficient increases by changing the angle of attack up to 16 degrees. Furthermore, increase in angle of attack leads to stall. However, at mach number 0.5 the lift coefficient increases up to 12 degrees beyond that stalling occurs. Thus from the results it is clear that naca0012 is suitable only for low speed flows.

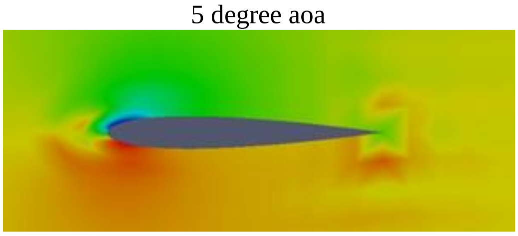
For Mach 0.1



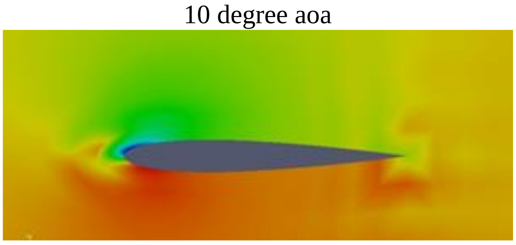
0 degree aoa



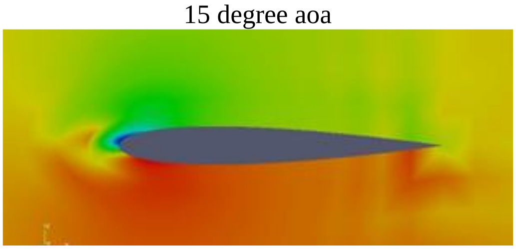
5 degree aoa



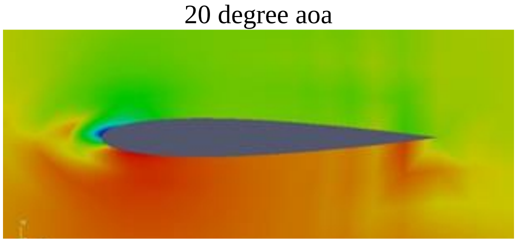
10 degree aoa



15 degree aoa



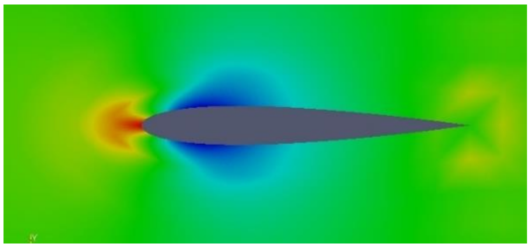
20 degree aoa



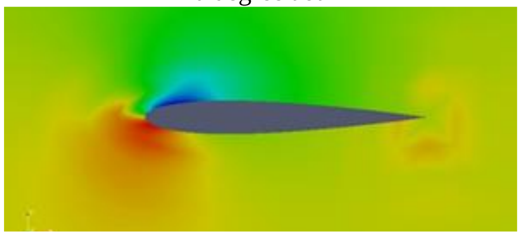
25 degree aoa

Fig 4 pressure contour at mach 0.1 over the airfoil

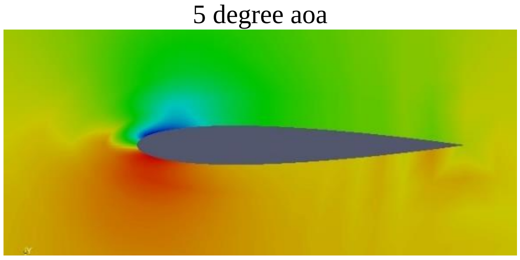
For mach 0.5



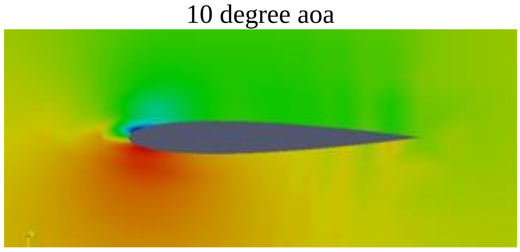
0 degree aoa



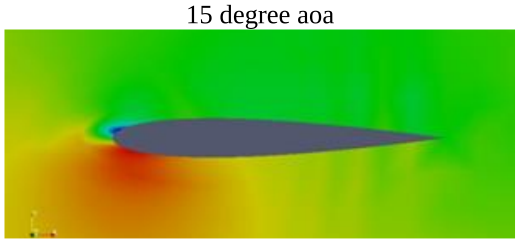
5 degree aoa



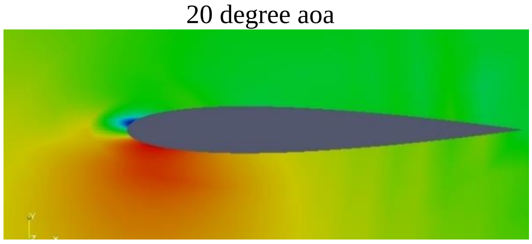
10 degree aoa



15 degree aoa



20 degree aoa



25 degree aoa

Fig.5 Pressure contour over the airfoil at mach number 0.5

ACKNOWLEDGMENT

The author would like to thank Mr. A. Pon Maa Kishan, President of Society of Aeronautical Engineers for his motivation to carry out this work.

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