ISSN (O): 2393-8609

International Journal of Aerospace and Mechanical Engineering

Volume 5 – No.2, April 2018

AERODYNAMIC CHARACTERISTICS OF DIAMOND SHAPED AIRFOIL AT SUPERSONIC SPEED

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ABSTRACT

In this paper, I investigated the flow over a "Diamond shaped Airfoil" in terms of Lift, Drag, L/D ratio and flow velocity over it at supersonic speed, Mach 2 at different angle of attacks. Diamond Airfoils experience lower drags at supersonic speeds as compared to circular Airfoils. Diamond airfoils can be proved to be more efficient Airfoils than other Airfoils at supersonic speeds. Hence it is very important to study the Aerodynamic characteristics of the Diamond Airfoil as it can solve the problem of high drag at supersonic speeds. I compared the flow parameters between three diamond shaped airfoil having different geometries by using one of the best computational fluid simulation software, ANSYS Workbench and Catia V5 for modelling.

Keywords

Diamond Airfoil, Supersonic Airfoil, Hypersonic Airfoil, Aerodynamic, Airfoil, Angle of Attack, Lift and Drag, Mach number, computational fluid dynamics, CFD, ANSYS simulation.

1. INTRODUCTION

An Airfoil (American English) or Aerofoil (British English) is the shape of a wing, blade (of a propeller, rotor, or turbine), or sail (as seen in cross-section).

An airfoil-shaped body moved through a fluid produces an aerodynamic force. The component of this force perpendicular to the direction of motion is called lift. The component parallel to the direction of motion is called drag.

The lift on an airfoil is primarily the result of its angle of attack and shape. When oriented at a suitable angle, the airfoil deflects the oncoming air (for fixed-wing aircraft, a downward force), resulting in a force on the airfoil in the direction opposite to the deflection. This force is known as aerodynamic force and can be resolved into two components: lift and drag. Most foil shapes require a positive angle of attack to generate lift, but cambered airfoils can generate lift at zero angle of attack. This "turning" of the air in the vicinity of the airfoil creates curved streamlines, resulting in lower pressure on one side and higher pressure on the other. This pressure difference is accompanied by a velocity difference, via Bernoulli's principle, so the resulting flow field about the airfoil has a higher average velocity on the upper surface than on the lower surface. The lift force can be related directly to the average top/bottom velocity difference without computing the pressure by using the concept of circulation and the Kutta-Joukowski theorem.

The primary purpose of an airfoil is to produce lift when placed in a fluid stream which of course experience drag at the same time. In an aircraft, lift on the wing surfaces maintains the aircraft in the air and drag absorbs all the engine power necessary for forward motion of the craft. In order to minimize drag, an airfoil is a streamlined body. The ratio of lift to drag gives a measure of the usefulness of an airfoil as a wing section of an aircraft. The higher this ratio the better the airfoil since it is capable of producing high lift at a small drag penalty. The ratio of lift to drag is expressed as L/D ratio or CL/CD ratio and can be determined by computational methods. If this ratio is high, then the airfoil can be used to produce useful lift which makes the aircraft to fly. An aircraft has a high L/D ratio or CL/CD ratio if it produces a large amount of lift or a small amount of drag.

2. METHODOLOGY

Theoretical considerations

2.1.2 Thin Airfoil Theory

Thin airfoil theory is a simple theory of airfoils that relates angle of attack to lift for incompressible, inviscid flows. It was devised by German-American mathematician Max Munk and further refined by British aerodynamicist Hermann Glauert and others in the 1920s. The theory idealizes the flow around an airfoil as two-dimensional flow around a thin airfoil. It can be imagined as addressing an airfoil of zero thickness and infinite wingspan. Thin airfoil theory was particularly notable in its day because it provided a sound theoretical basis for the following important properties of airfoils in two-dimensional flow:

1. On a symmetric airfoil, the center of pressure and aerodynamic center lies exactly one quarter of the chord behind the leading edge.
2. On a cambered airfoil, the aerodynamic center lies exactly one quarter of the chord behind the leading edge.
3. The slope of the lift coefficient versus angle of attack line is **2π** units per radian.

As a consequence, the section lift coefficient of a symmetric airfoil of infinite wingspan is:

$$Cl = 2*\pi*\alpha$$

where **Cl** is the section lift coefficient

α is the angle of attack in radians, measured relative to the chord line.

3. COMPUTATIONAL METHOD

Modelling

A schematic of the geometric model of the Airfoil used for the analysis is shown in figure:

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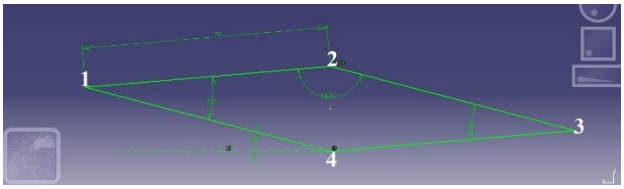


Airfoil Name	Angle 1	Angle 2	Angle 3	Angle 4
A	20	160	20	160
B	10	170	10	170
C	05	175	05	175

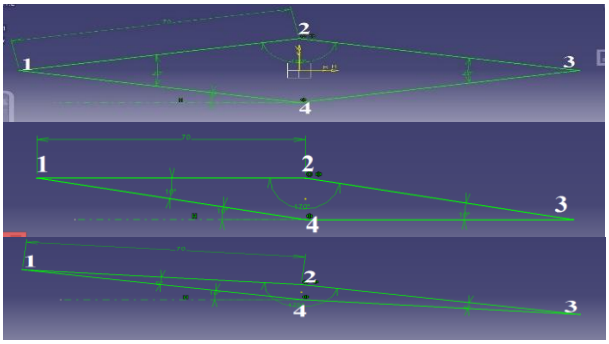
To make this geometry, I used CatiaV5 using generative shape design module. The cord of the Airfoil is 140.034mm. I varied the geometry of the Airfoil by changing the angles of the Diamond geometry--the upper and lower angles namely 160 deg, 170 deg, 175 deg.

Simulation

The simulation was done using the FLUENT over the ANSYS Workbench. I used the pressure based solver and time steady planar method. The model used was Sparlat-Allamaras(1eqn.). The fluid used is air and the material of the airfoil is the Aluminum. The ell zone conditions were kept as default. Now regarding the Boundary conditions, the inlet was set as a velocity inlet type with the value of Mach 2(686m/s). The outlet was set as pressure outlet with the default values. The solution methods were- the simple scheme for pressure velocity coupling. Least squares cell based, Second order upwind momentum for the spatial discretization. Monitors for the CL and CD plot. Hybrid settings for the solution initialization. Did run the iterations till the solution got converged.

4. RESULTS AND DISCUSSIONS

I analyzed the flow over three types of Diamond shaped airfoils. I took the airfoils changing the upper and lower angles-160 deg.170 deg.175 deg. with a negligible change in chord length. As shown in the figures



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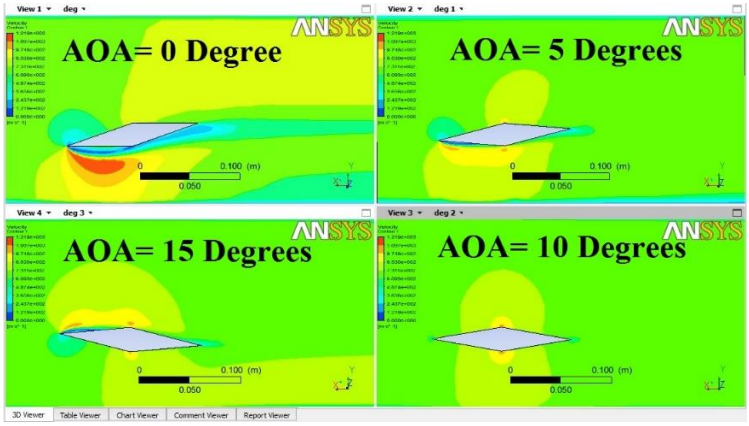
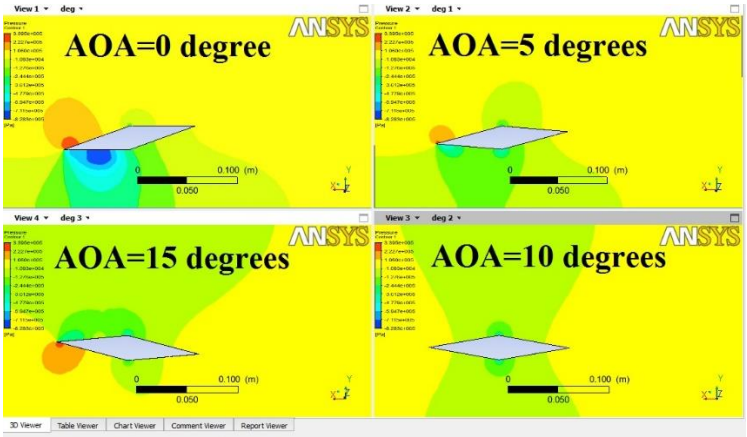
4.3.1 For the first Airfoil (A)

The results for the Airfoil with geometric angles as 160,160,20,20 deg. and cord length 140.034 mm at Mach 2 for different angles of attack, the pressure and velocity contours were as given

The velocity contours were as shown



The pressure contours were as shown:



AOA	Maximum Velocity	Maximum Pressure
0 degree	1218.52 m/s	325500 Pa
5 degrees	1084.45 m/s	300388 Pa
10 degrees	1184.14 m/s	180605 Pa
15 degrees	1130.56 m/s	339507 Pa

The **AOA** vs **CL** plot for the given Airfoil is:

4.3.2 For the second Airfoil (B)

The results for the Airfoil with geometric angles 170,170,10,10 deg. and cord length 140.034mm for inlet velocity at Mach 2. The pressure and velocity contours are:

The velocity contours for the given Airfoil

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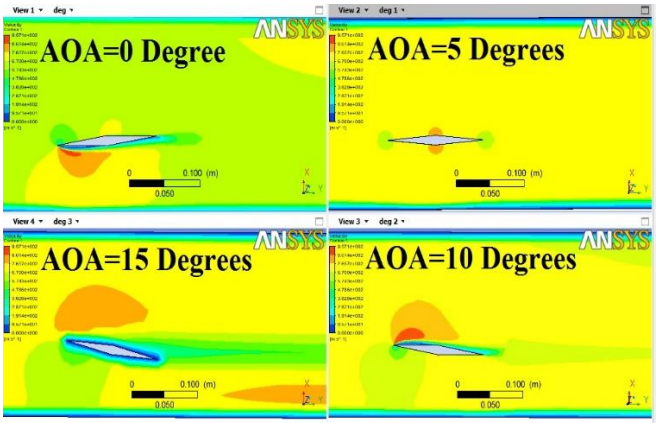
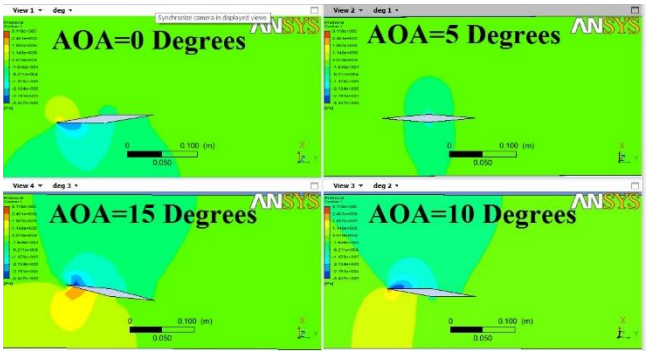
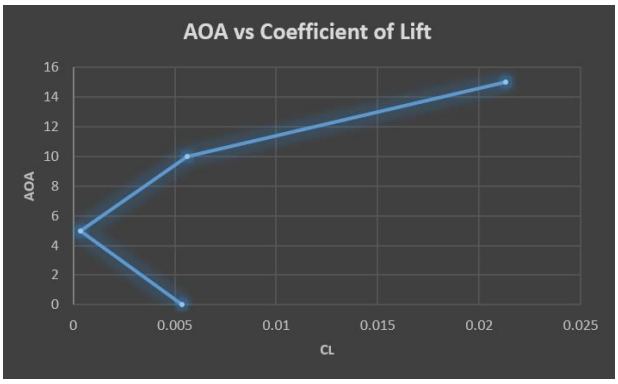


The Pressure contours for the given Airfoil:



AOA	Maximum Velocity	Maximum Pressure
0 degree	903.284 m/s	260978 Pa
5 degrees	884.967 m/s	158595 Pa
10 degrees	957.113 m/s	311779 Pa
15 degrees	858.574 m/s	246442 Pa

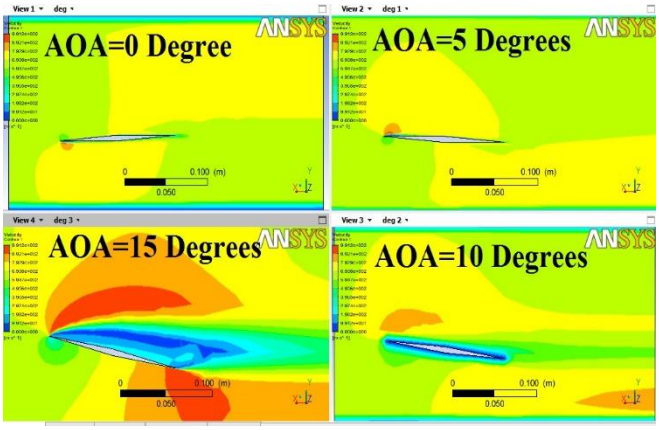
The **AOA** vs **CL** plot for the given Airfoil is:



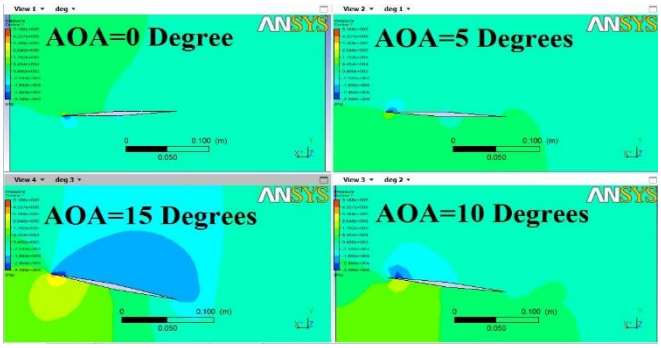
4.3.3 For the third Airfoil (C)

The results for the Airfoil with geometric angles 175,175,5,5 deg. and cord length 140.034mm for inlet velocity at Mach 2. The pressure and velocity contours are:

The velocity contours for the given Airfoil

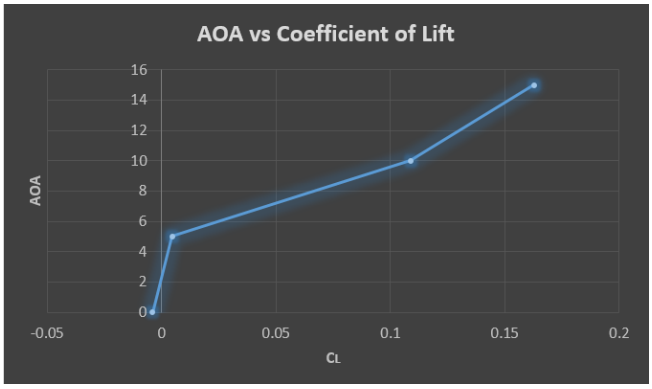


The Pressure contours for the given Airfoil



AOA	Maximum Velocity	Maximum Pressure
0 degree	918.524 m/s	264975 Pa
5 degrees	929.793 m/s	281711 Pa
10 degrees	852.456 m/s	249208 Pa
15 degrees	991.170 m/s	518529 Pa

The **AOA** vs **CL** plot for the given Airfoil is:



5. CONCLUSIONS

Comparison of the data of the three Airfoils--A, B, C.

Airfoil for Min. and Max. Velocity at various AOA:

AOA(deg)	Max. Velocity(m/s)	Min. Velocity(m/s)
00	A (1218.52)	B (903.284)
05	A (1084.45)	B (884.967)

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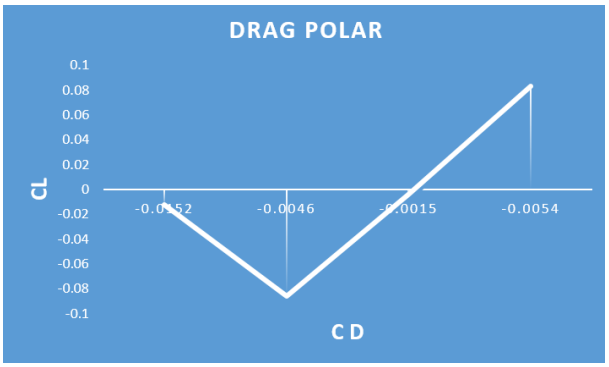
10	A (1184.14)	C (852.456)
15	A (1130.56)	B (858.574)

Airfoil for Min. and Max. Pressure at various AOA:

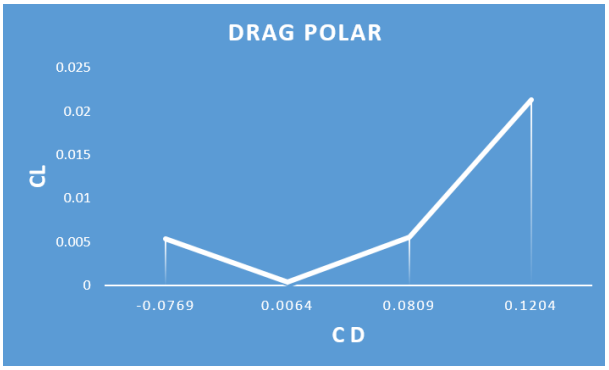
AOA(deg)	Max. Pressure(Pa)	Min. Pressure(Pa)
00	A (325500)	B (260978)
05	A (300388)	B (158595)
10	B (311779)	A (180605)
15	C (518529)	B (246442)

5.1 DRAG POLAR FOR THE AIRFOILS:

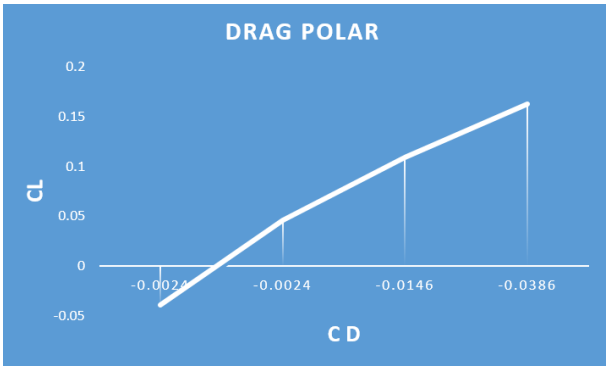
Airfoil **A**



Airfoil **B**



Airfoil **C**



5.2 VERDICT

It is clear from the above comparison tables that, the Airfoil A gives the best result and the Airfoil B gives the worst result among these Airfoils, when it comes to the matter of speed. But, when it comes to the matter of Lift and Aerodynamic efficiency, then according to the Drag polar of the Airfoils, Airfoil C gives the best result.

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