Shockwave Study on The Wings NACA

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Submission date: 09-Apr-2023 06:29AM (UTC+0700)
Submission ID: 2059222040
File name: Shockwave_Study_on_The_Wings_NACA.pdf (499.31K)
Word count: 2720
Character count: 12573

https://doi.org/10.51630/ijes.v2i1.10

SHOCKWAVE STUDY ON THE WINGS NACA 0012, NACA 64 - 206, AND NASA SC (2) - 0706 WITH $\Lambda = 15^{O}$ AT 0.85 MACH NUMBER

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(Received: 15 January 2021; Accepted: 25 February 2021; Published on-line: 01 March 2021)

ABSTRACT: Airfoil is used as a basic form on aircraft wings. Airfoil on the wing of the aircraft is used to produce lift that will lift the fuselage into the air. Lifting force results from the difference in pressure between the upper surface and the lower surface of an aircraft wings. In high speed flights shockwave will occur at certain parts of the wing which will adversely affect the aerodynamic performance of the wing. Wing aerodynamic performance at high speeds can be improved in various ways, one of which is by giving a angle to the wing span called a swept angle. This study will use 3D CFD simulation methods using Ansys Fluent. The airfoil used are NACA 0012, NACA 64-206, and NASA SC (2) -0706 with a chord length of 1 m, AR = 5, and $\lambda = 1$ with backward swept angle $\Lambda = 15$ °. Free stream flow is air flowing with Mach Number = 0,85 at sea level and steady conditions. Based on the simulation results, shock occurs on the upper and lower surfaces for NACA 0012 with Cl = 0 due to symmetric airfoil, whereas shock occurs only on the upper surface for NACA 64-206 and NASA SC (2) - 0706 with a Cl / Cd value of 18.55 (NACA 64-206) and 20.78 (NASA SC (2) - 0706). This simulation also provides a visual representation of Mach Number contour plots in the middle stretch (Midspan) of the wing and Cl and Cd data.

KEY WORDS: Airfoil, Mach Number, Shockwave, Swept angle.

1. INTRODUCTION

In an airplane, the wing function is vital, providing lift force. The force generated is due to the difference in pressure on the upper and lower surfaces of the wing. The difference in pressure and the characteristics of the fluid flowing on the wing depends on the profile of the airfoil used in the wing construction [1].

Airfoil is an aerodynamic structure that is widely used in both aircraft wings and fluid engines such as pumps, compressors, and turbines. The difference in pressure between the top and bottom of the airfoil causes the aircraft to gain lift. At transonic flight velocity, the flow on the upper surface of the wing will form a shockwave [2].

At transonic flight velocity, the flow on the upper surface of the wing will form a shockwave[3]. This wave drag phenomenon can be controlled by slowing down the shockwave formation on the wing, this can be done by providing a swept angle at the leading edge of the wing, the swept angle can be forward or backward and will have the same effect, wing configuration such as this is called a swept wing [4].

The formation of a shockwave reduces the flow velocity significantly, preventing the aircraft from accelerating further to reach sonic and supersonic speeds A decrease in flow velocity will also reduce the pressure difference on the upper and lower surfaces of the wing to be much smaller, the resulting lift will also decrease. This phenomenon is often referred to as wave drag [4-8] [5-9].

The shockwave formation on the wing can be delayed by providing a swept angle and using a thinner airfoil so that local flow acceleration does not reach the sonic (M = 1). This wing configuration is called a swept wing. Swept wings are often used on commercial aircraft operating at high flying speeds such as the Boeing 757, Boeing 747, Airbus A350 and Airbus 380 as well as on military aircraft with supersonic capabilities such as the Sukhoi SU-35 and McDonnell Douglas F-4 Phantom [3, 10-12].

A research conducted on an analysis of the NACA 0012 airfoil wing by varying the backward swept angle of 0, 15 and 30^{o} and variations of the angle of attack of 8 and 15^{o}[1]. The performance comparisons shows that at α = 15^{o} is a very significant increase in Cd for each Λ so that the ratio of Cl/Cd will decrease drastically about 1/3 of the value when $\alpha = 8^{\circ}$.

Yen (2011) conducted an experiment to analyze the aerodynamic characteristics and shedding of the wing with backward swept [1]. Reynolds's number = 1 x 10^{5} where $\Lambda = 15^{\circ}$. In the leading edge bubble and separation bubble, the value of Cm increases in proportion to the increase in α until a point experiences a stall. The increasing drag in the bubble burst area causes the value of to reach the maximum. Cm increases in the turbulent separation area due to the large increase in drag.

Abdullah et al (2017) conducted a numerical study on military aircraft airfoils designed for compressible flow [12]. The airfoils used are NACA 0003, NACA 0012, NACA 64-206, and NACA 64-210. The sharp leading edge construction causes the air flow above the thin airfoil to be more susceptible to flow separation, leading to higher stall and drag at low angles of attack.

The high velocity fluid flow flowing through the wing will creates a shockwave caused by compressibility of air at sonic velocity. This shockwave can later cause a wave drag phenomenon that is detrimental to the aircraft. Giving a swept angle can reduce the amount of chord wise flow so that it can increase wing performance at high speeds and prevent wave drag phenomena.

In this study, the NACA 0012, NACA 64 – 206 and NASA SC (2) – 0706 [13, 14], airfoil was used. The selection of this airfoil type is based on the high speed flight concept where a thin airfoil, a sharper leading edge, and a special contour is required in order to have optimal performance at high speeds.

2. RESEARCH METODOLOGY

In this research, the simulation process begins with drawing geometry and meshing. The geometry is designed using three Solid works airfoil designs with a backward swept angle of 15^{o}. The meshing method used is a tetrahedron (Figure 1) with high density in the area around the wing, which will become more tenuous if it is away from the wing. This meshing method is used to lighten the computational burden and get fairly accurate results.

After meshing the model, export it to the FLUENT CFD software. The desired result will show the shockwave condition when the wing is given a swept angle which is generally seen from the Mach number value of the flow on the upper and lower surface. Different airfoil types will cause differences in the local Mach number values under the same free stream conditions. The simulation results will also show Cl and Cd for each airfoil.

Indonesian Journal of Engineering and Science, Vol. 2, No. 1, 2021 *Puspitasari and Kiat*
https://doi.org/10.51630/ijes.v2i1.10

The wing geometry was drawn using Solid work software. The geometry is designed based on data obtained from the central airfoil database (Figure 2). Each airfoil will be used as a wing with a chord length of 1 m and a 5 m wingspan and then a backward swept angle (Λ) of 15o is given. Figure 3 shows an example of a wing design.

The CFD simulation method is used by first determining the solver, model, and input boundary conditions, namely: i) the solver used is pressure based, assuming compressible and steady flow; ii) the viscous model used is Spalart-Allmaras; iii) boundary condition is free stream flow at sea level with Mach Number 0.85; and iv) free stream data are shown in Table 1.

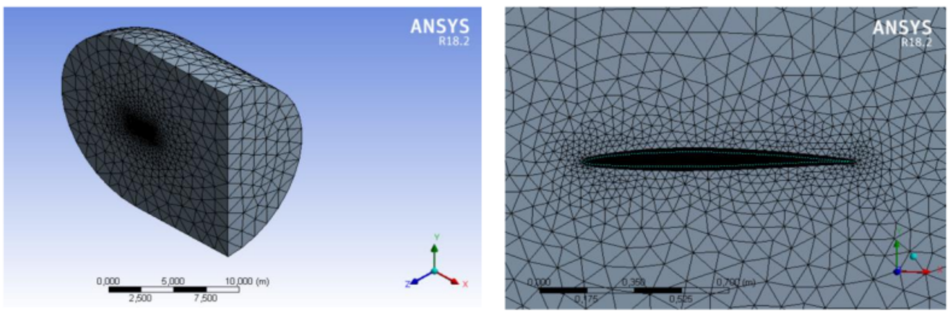

Fig. 1. Meshing tetrahedron in geometry

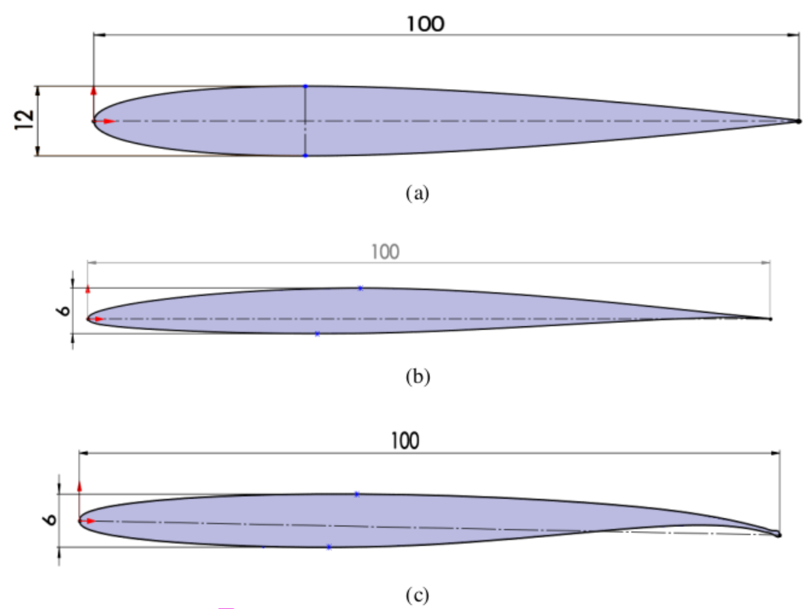

Fig. 2. The geometry of NACA 0012 (a), NACA 64 -206 (b), and NASA SC (2) - 0706

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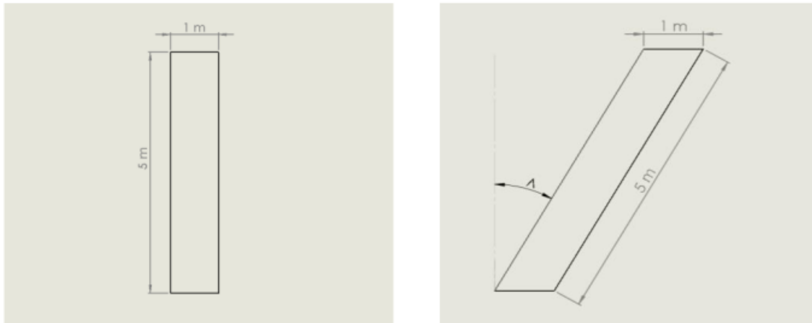


(a) (b)

Fig. 3. Examples of straight wing geometry designs (a), and with angle Λ (b)

Table 1. Free stream data

Parameters	value
Pressure (P)	101.325 Pa
Density (ρ)	1.225 kg/m^3
Temperature (T)	15 ^{o}C or 288.15 K
Sound Velocity (*a*)	340.3 m/s
Viscosity (μ)	1.789 x 10^{-5} kg/m/s
Kinemtatic Viscosity (v)	1,460 x 10^{-5} m^2/s

3. RESULTS AND DISCUSSION

The CFD simulation results obtained the Mach number contours for each airfoil on the midspan section with the XY plane orientation. A shockwave starts to form when the Mach number of the local flow is equal to 1 which is called the sonic velocity. The position for taking the Mach number data is located on the midspan for each variation Λ using equation (1).

$$0.5 \times b \cos \Lambda \tag{1}$$

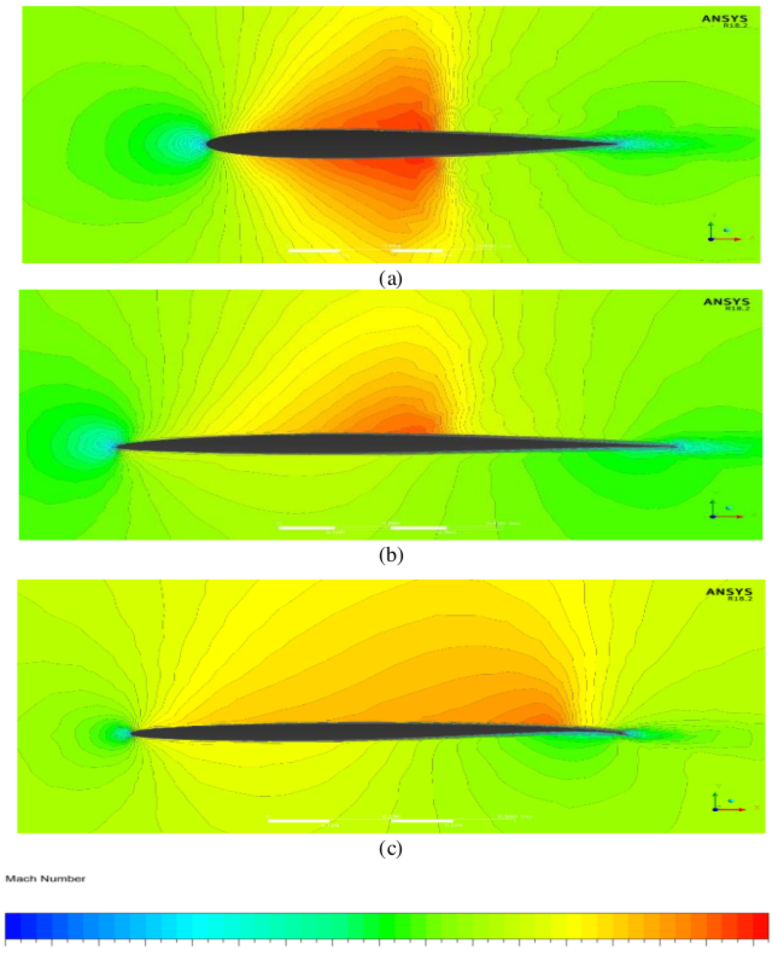
Where b is the length of the wingspan (m) and Λ is the swept angle. To get the flow velocity from the Mach number value, equation (2) can be used.

$$v = M.a \tag{2}$$

Where v is the velocity of flow (m/s), M is the Mach number and is the speed of sound at a certain height (m/s).

The Mach number value of the flow around the airfoil is shown in Figure 4. The airfoil is in the flow with transonic velocity so that there will be a shockwave due to the compressibility effect of the fluid and the decrease in the Mach number to subsonic after the flow passes through the shockwave. The difference in contour fields is seen in NACA 0012 with NACA 64-206 and NASA SC (2) -0706 as expected because NACA 0012 is a symmetrical airfoil.

Signs of boundary layer flow separation begin to appear on the wing surface immediately behind the shockwave when the shock condition is strong enough. The flow in front of the shockwave reaches supersonic then returns to subsonic after passing through the shockwave. In NACA 0012 shock occurs on the upper and lower surfaces of the wing, this is because NACA 0012 is a symmetrical airfoil so that the flow velocity on the upper and lower surfaces is the same.

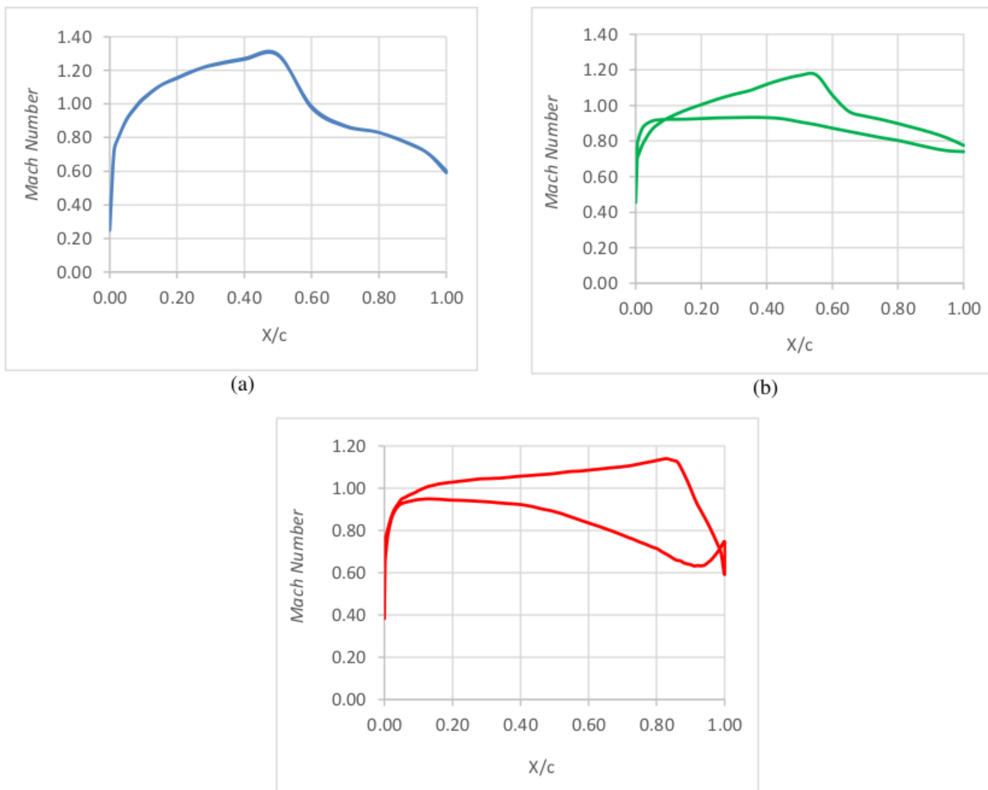


(d)

Figure 4. Mach number contour for airfoils (a) NACA 0012, (b) NACA 64 - 206, (c) NASA SC (2) – 0706, (d) Legend

Whereas in NACA 64-206 and NASA SC (2) -0706 the shock only occurs on the upper surface because the airfoil is not symmetrical so that the velocity on the lower surface is lower and not enough to form a shock, but if the free stream speed is increased shock can form on the lower surface airfoil.

The supercritical profile of the NASA SC (2) -0706 airfoil causes the shock to form behind the trailing edge, this can keep the pressure difference between the top and bottom surfaces high so that it has better lift than the other two airfoils. The shock strength can be seen from the difference in the Mach number between the front and rear shockwave. The difference in Mach number (M) can be seen from the M graph against the location versus the chord line (X/c) in Figure 5 where the shock causes a sudden significant decrease in M value, the greater the decrease in M value, the stronger the shockwave.



(c)-

Figure 5. Graph of M against X / c for airfoil NACA 0012 (a), NACA 64 - 206 (b), NASA SC (2) - 0706 (c).

The shockwave can be delayed and attenuated by the use of a thinner airfoil and the use of a swept angle. The swept angle will divide the flow into chord wise and span wise components as a result, the chord wise velocity which has a direct impact on shock formation can be reduced. Chord wise speed also affects the lift so that the use of a swept angle that is too large should be avoided because it can reduce lift. The lift and drag coefficient values for the three airfoils can be seen in Table 2 below.

Table 2. Value of Cl and Cd for each airfoil

	NACA 0012	NACA 64 – 206	NASA SC(2) - 0706
C_l	0.000137	0.1929	0.3575
C_d	0.0261	0.0104	0.0172
C_l / C_d	0,00525	18,55	20,78

4. CONCLUSION

The simulation results on NACA 0012 show that there are supersonic speeds on the upper and lower surfaces due to the symmetrical airfoil. For NACA 64-206 airfoil there is a weaker shock due to the thinner airfoil and NASA SC (2) -0706 shock occurs close to the trailing edge because the supercritical profile of the airfoil shock strength can be seen from the difference in mach number between the front and rear shockwave.

The C_l value in NACA 0012 is very small and is close to 0 in all due to the very small pressure difference between the upper and lower surfaces of the symmetrical airfoil, for NACA 64-206 it has a C_l value of 0.19 and for NASA SC (2) -0706 it has a value C_l of 0.36. Therefore, based on aerodynamic performance, the best C_l value of the airfoil among the three is the NASA SC (2) -0706 airfoil because this airfoil has the highest C_l than other airfoils.

ACKNOWLEDGEMENT

The authors would like to convey their great appreciation to Universitas Sriwijaya for supporting this research.

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