Numerical analysis of the subsonic flow around NACA 0015

BOUBABOURI Benaissa, MADANI YSSAAD Habib

***Abstract*—** The objective of this study is to study the subsonic flow around a wing. In this article, a wing NACA0015 is studied. Simulations are performed at a Mach number of 0 .3 up to 1.0 at three angles of attack (AOA) of 2, 4 and 6 degrees. The numerical tool used (Fluent) in version 15 is based on the finite volume method for the resolution of the Reynolds average equations and the energy equation.

***Keywords*—** Subsonic, NACA0015, Drag, Pressure, Critical, Divergence, Boundary layer, compressibility.

I. Introduction

THE aerodynamics of the wing profiles is a very recent science in the field of mechanics, since the first research work goes back to the twenty first centuries. The aerodynamics of a wing profile moving in relation to its surrounding environment has long attracted the interest of researchers both fundamentally and applied, and both numerical and experimental [1, 7].

Jet aircraft fly in the lower transonic regime, also called high subsonic regime. It is therefore interesting to examine more closely the flow phenomena that occur at these speeds and their impact on the design and operation of the aircraft. To be able to justify a particular aircraft design, we need to have a thorough understanding of the flow phenomena and how we can influence them to meet the requirements defined by the customers while satisfying the constraints of the aviation authorities.

Initially, the flow is subsonic throughout the body. Even if the flow is accelerated on the body, nowhere in the flow domain does the Mach number increase beyond $M_{\infty} = 1$. When the Mach number increases beyond the critical Mach number, Mcr, a supersonic domain begins to form. Due to the shape of this body, this supersonic area is likely to form on the part of the body that has the highest curvatures, which is near the leading edge of the body.

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In this case, a supersonic flow pocket is formed on the upper and lower sides of the body. Aerodynamics shows us that the flow behind the body must be close to the speed of the free flow (which is subsonic). Therefore, the flow in the supersonic pocket must be slowed down. In the present example, this deceleration occurs quite abruptly by means of a normal shock wave almost perpendicular to the surface of the bearing surface. In this article, we studied these phenomena on a wing NACA0015, this wing has been used in many applications, including a plane drift (the empennage), underwater fins, rotating fins and some fixed wings.

II. Numerical Procedure

A. Import Geometry

We import the points defining the profile. These points are the coordinate values given by a series of points calculated from the NACA 0015 formula.

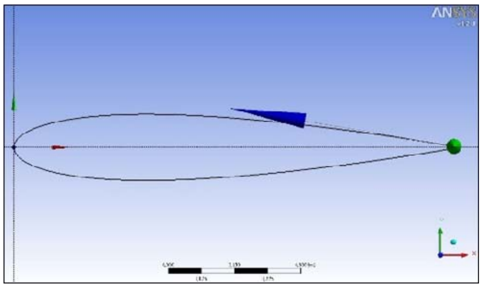

Fig. 1 Profile NACA0015

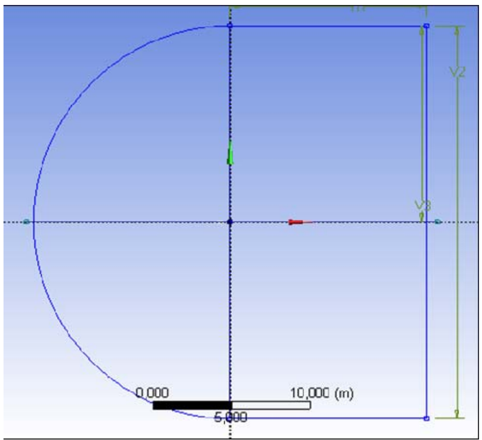

Fig. 2 Calculation domain

B. Mesh

A complete mesh around a profile NACA0015 and more refined near the surface of the profile to detect the details of the boundary layer, are shown in Figures 3 and 4. As you will be completing this form online, you do not need to fill out a hard-copy form. Do not include a copyright statement anywhere on your paper.

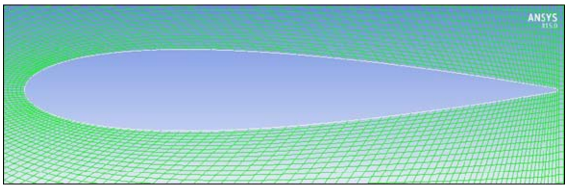

Fig. 3 Mesh close to the profile

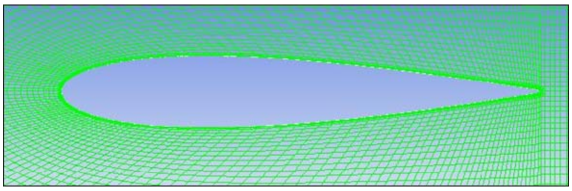

Fig. 4 Refined mesh

C. Validation of Computer Code

To validate simulation results, the lift coefficient and the drag coefficient of profile NACA0015 compared with experimental data results [8] present in figures 5 and 6.

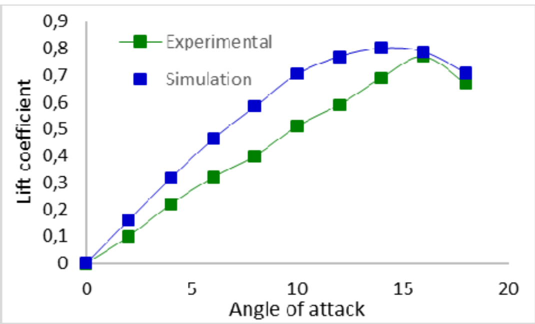

Fig. 5 Lift coefficient comparison between experimental data and simulation data

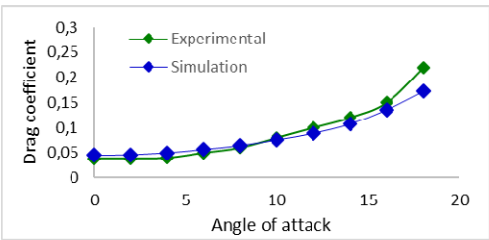

Fig. 6 Drag coefficient comparison between experimental data and simulation data

The lift coefficient increases linearly with the angle of attack. It can also be noted that there is a loss of lift at an angle of attack of about 16 °, this phenomenon known as stall. A good agreement with the experimental data.

III. Presentation And Interpretation Of The Results

A. Pressure Coefficient

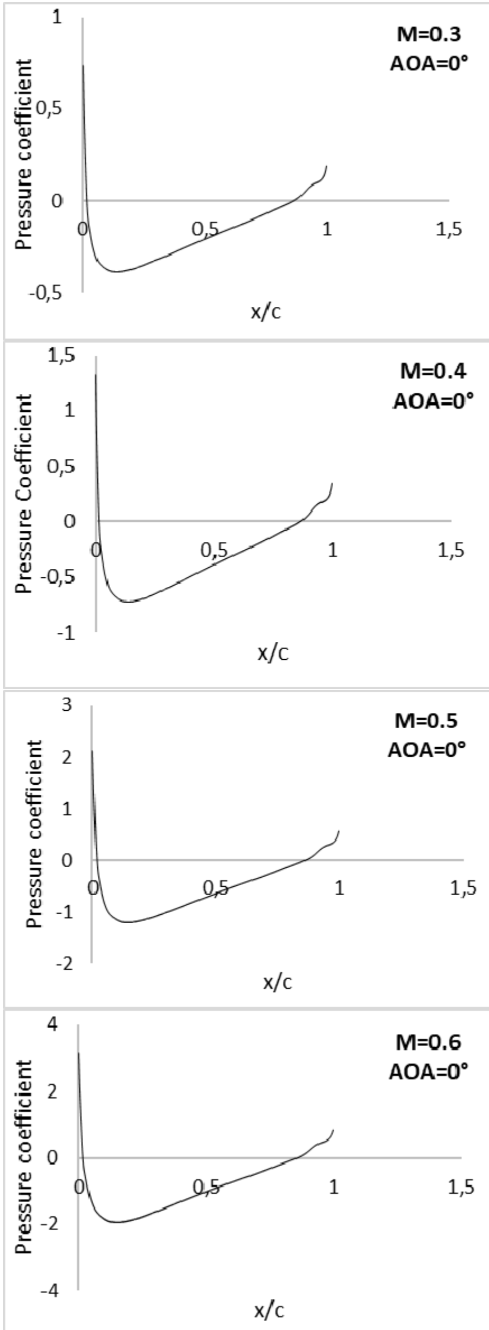

Fig. 7 Pressure Coefficient Distribution for Different Mach Number Values; AOA=0°

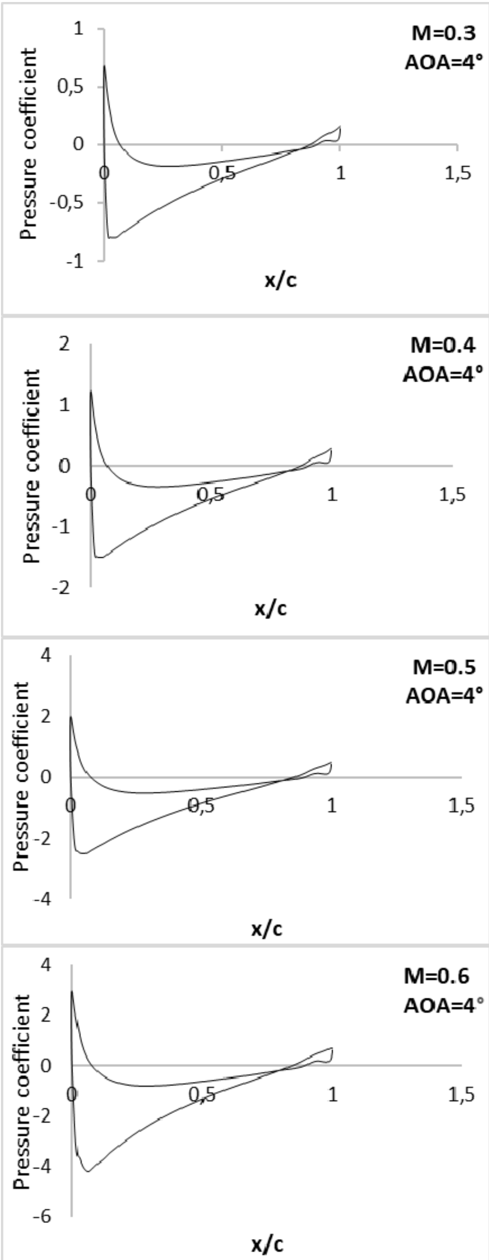

Fig. 8 Pressure Coefficient Distribution for Different Mach Number Values; AOA=4°

If the flight speed U□ is small compared to the speed of sound in the medium, the medium in which the body moves can be considered as incompressible. Indeed, the compressibility has practically no effect when the perturbation P' of the pressure generated by the movement of the body is very small compared to the pressure P□ in the undisturbed medium:

$$P'/ P_{\square} << 1 \qquad (1)$$

The pressure coefficient depends essentially on the coordinates of the point where P is determined, the shape of the body and its position in the flow. an increase of M□ will always result in a further increase of Cp, but it can be seen from Figures 7 and 8 that the dependence of P on M□ is relatively low for subsonic velocities, the effects of compressibility of the medium will be felt for a weak M□. Therefore, the effect of medium compressibility will be stronger for a higher M□ in the incoming flow. and for stronger disturbances caused by the body for small M□.

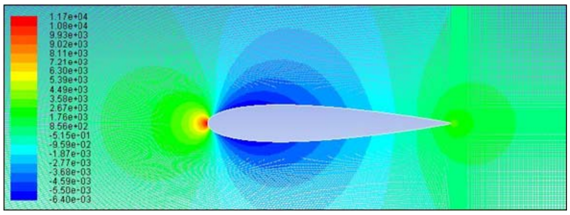

Fig. 9 Pressure contour; M□=0.4; AOA=0°

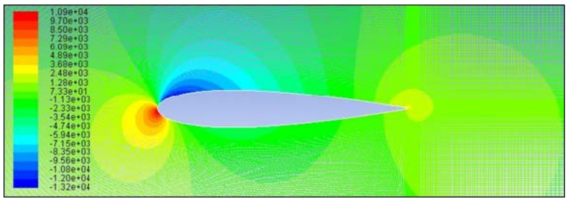

Fig. 10 Pressure contour; M□=0.4; AOA=4°

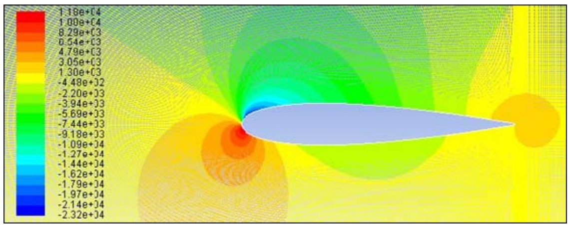

Fig. 11 Pressure contour; M□=0.4; AOA=6°

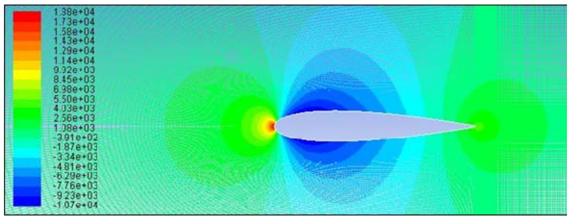

Fig. 12 Pressure contour; M□=0.5; AOA=0°

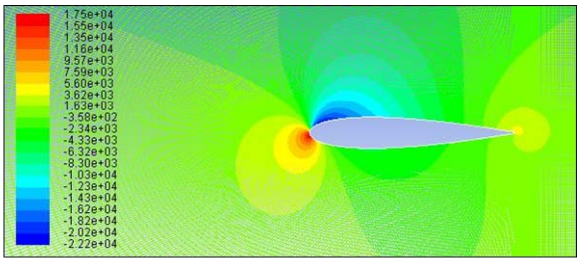

Fig. 13 Pressure contour; M□=0.5; AOA=4°

Figure 14 shows the evolution of the pressure coefficient Cp on the profile NACA0015 for different values of the angle of attack AOA = 4°, 6° and 12° with M□ = 0.4.

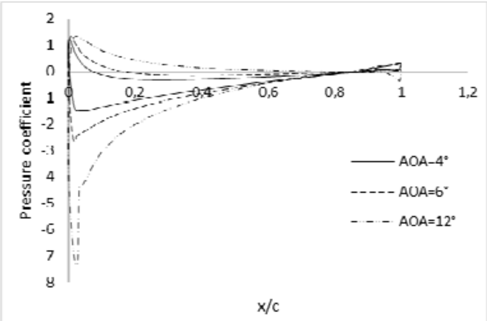

Fig. 14 Pressure coefficient distribution for different values AOA with M_{∞} = 0.4

The lift is result of the distribution of the pressure on the extrados and the intrados of the profile. It is represented by the distribution of the pressure coefficient Cp on either side of the profile. So the lift is represented by the area between the two curves of Cp on the extrados and the intrados. Then from Figure 14 we see that when the angle of attack increases, the lift increases.

B. Critical Mach Number

From Figures 9,10,11,12 and 13, it can be seen that when the aircraft is flying at a subsonic speed, there is an area around the upper part of the wing (the extrados) where the pressure is minimal. where M (local Mach number) is the maximum ($M > M_{\infty}$) (see figures 15, 16 and 17). We increase M_{∞} until we reach the exact value such that the local Mach number at the minimum pressure point is equal to 1, that is to say that M = 1. When this happens, the Mach number M_{∞} is called the critical Mach, denoted by Mcr.

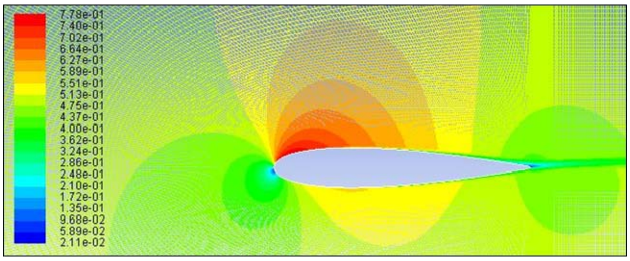

Fig. 15 Mach number contour; M_{∞}=0.5; AOA=4°

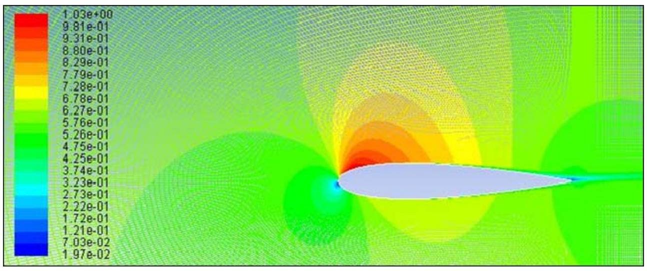

Fig. 16 Mach number contour; M_{∞}=0.6; AOA=4°

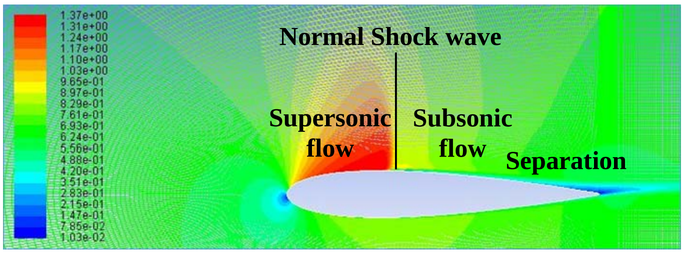

Fig. 17 Mach number contour; M_{∞}=0.7; AOA=4°

By definition, the critical Mach number of an aircraft is the lowest Mach number at which the airflow on a point on the aircraft reaches the speed of sound.

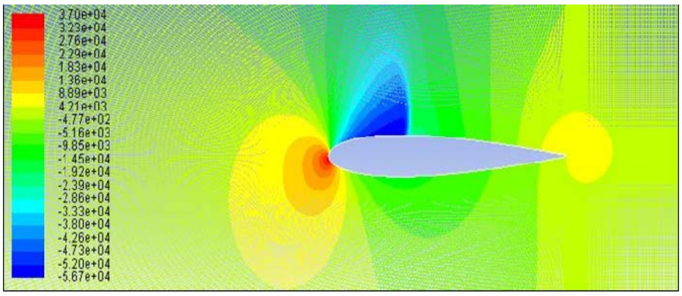

Fig. 18 Pressure contour; M_{∞}=0.7; AOA=4°

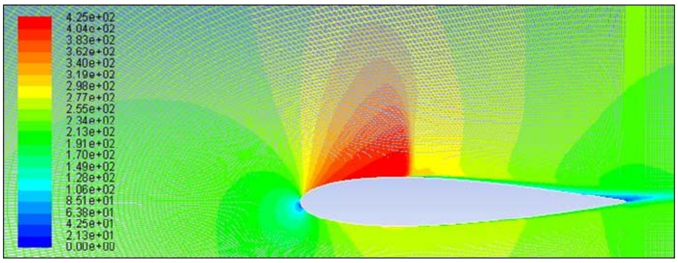

Fig. 19 Velocity contour; M_{∞}=0.7; AOA=4°

from this study, Figure 20 illustrates the changes that occur in physical parameters when a flow passes through a normal shock wave.

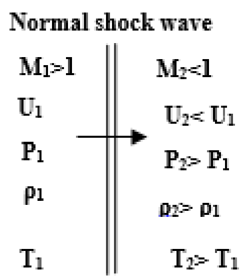

Fig. 20 Flow through normal shock waves

One of the most important problems in high speed aerodynamics is the determination of the critical Mach number of a given bearing surface, so that M_{∞} values slightly above Mcr, the supersonic airflow at above the wing is reduced to the flow of subsonic air and shock waves are generated, which causes turbulence followed by instability, therefore the flow undergoes a sudden increase in pressure, density and of the temperature. The energy associated with these changes is extracted from the total energy of the flow to reduce the velocity behind the wave front (see Figures 17, 18 and 19). Together, these modifications are considered to be a sharp increase in drag in particular and may be accompanied by significant changes in aircraft trim and stability and control characteristics (see Figures 21 and 22).

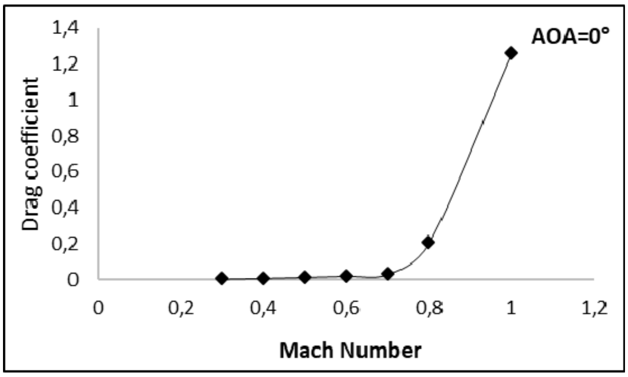

Fig. 21 Drag coefficient for different Mach number; AOA = 0 °

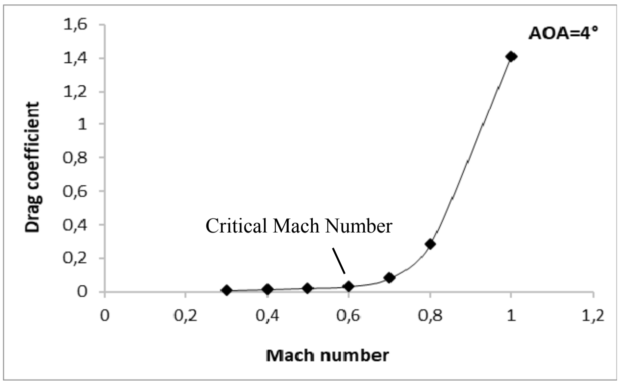

Fig. 22 Drag coefficient for different Mach number; AOA = 4 °

IV. Conclusion

The drag associated with high subsonic velocities results from shock waves that terminate the supersonic flow zones on the wing. At first there is a slow increase in drag (called drag creep). The local shock wave at the top of the wing creates a very strong negative pressure gradient above the boundary layer. When the shock wave is strong enough, this results in separation of the boundary layer, which creates an exponential increase in drag. The Mach number to which this occurs is called the Mach number of divergence of drag, M_{dd}. (The definition of the Mach number with divergence of drag is arbitrary).

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