Study of the Aerodynamic Characteristics of an Aerofoil in Accelerating Free Streams

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가속 유동장에서 발생하는 익형의 공력특성에 관한 연구

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Abstract

Many flight bodies are essentially imposed in gradually accelerating and decelerating free streams during taking-off and landing processes. However, the wing aerodynamics occurring in such a stream have not yet been investigated in detail. The objective of the present study is to make clear the aerodynamic characteristics of an aerofoil placed in the accelerating and decelerating free stream conditions. A computational analysis is carried out to solve the unsteady, compressible, Navier-Stokes equations which are discretized using a fully implicit finite volume method. Computational results are employed to reveal the major characteristics of the aerodynamics over the gradually accelerating aerofoil wings.

1. INTRODUCTION

Many flight bodies essentially experiences accelerating and decelerating free streams during their taking-off and landing processes[1,2]. A large number of researchers have been made to investigate the aerodynamic behaviors over an aerofoil at subsonic and supersonic speeds. Much has been learned from experimental and computational work with regard to the wing aerodynamics. Aerodynamic design of the aerofoil wing of modern commercial and combat aircrafts has become practically possible even though many unsolved problems remain associated with the shock boundary layer interactions over the wing, flow separation and detailed vertical behaviors at transonic and supersonic speeds.

Three different types of flows occur when the shock interacts with the turbulent boundary layer at high subsonic and transonic flows; type 1 interaction includes that weak shock thickens the boundary layer; type 2 includes that stronger shock locally separates the boundary layer; type 3 includes that very strong shock separates boundary layer to trailing edge[3,4].

The aerodynamic characteristics of an aerofoil placed in a gradually accelerating or decelerating free stream condition at subsonic speed have not been studied yet, while the wing aerodynamics of an abruptly accelerating or decelerating flow in the subsonic free stream conditions have been well-known[5].

The objective of the present study is to make clear the aerodynamic characteristics of an aerofoil in the gradually accelerating or decelerating subsonic free stream conditions. A computational analysis is carried out to solve the unsteady, compressible, Navier-Stokes equations which are discretized using a fully implicit finite volume method. Computational predictions are employed to reveal the major characteristics of the wing aerodynamics such as lift and drag over the aerofoil wing in the gradually accelerating or decelerating subsonic free stream conditions.

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2. NUMERICAL METHOD

2.1 Governing Equations

The governing equations are unsteady, compressible, implicit 2D Navier-Stokes equations. The resulting equations are as follows

$$\frac{\partial r}{\partial t}+\frac{\partial}{\partial x_i}(ru_i)=0 \tag{1}$$

$$\frac{\partial}{\partial t}(ru_i)+\frac{\partial}{\partial x_j}(ru_iu_j)=\frac{\partial}{\partial x_j}m\left(\frac{\partial u_i}{\partial x_j}+\frac{\partial u_j}{\partial xx_i}\right)-\frac{\partial}{\partial x_i}\left(\frac{2}{3}m\frac{\partial u_l}{\partial x_l}\right) -\frac{\partial p}{\partial x_i}+\frac{\partial}{\partial x_j}(-r\overline{u_i'u_j'}) \tag{2}$$

$$\frac{\partial}{\partial t}(rE)+\frac{\partial}{\partial x_i}(ru_iH)=\frac{\partial}{\partial x_i}\left[\left(x+\frac{m_t}{\mathrm{Pr}_t}\right)\frac{\partial T}{\partial x_i}+u_j(t_{ij})_{eff}\right] \tag{3}$$

The governing equations are discretized spatially implicit finite volume scheme. With respect to temporal discretization, explicit 4-stage Runge-Kutta time stepping scheme is used. Used turbulence model is standard k-e turbulent model which is a semi-empirical model based on model transport equations for the turbulence kinetic energy(k) and its dissipation rate(e).

2.2 Computational Domain and Grid System

Fig.1 shows the C-typed computational domain and 2D NACA0012 airfoil grid system used in the present study. 2D structured grid system is used to simulate the flow field of the wing and the grid points are about 15,000. The grids are dense around the wing surface to capture the shock-induced flow separation so that provide more accurate predictions of the flow field. The dimension of computational domain is setup at 20 times of chord length (c) toward upstream from the leading edge, 25 times of chord length toward downstream.

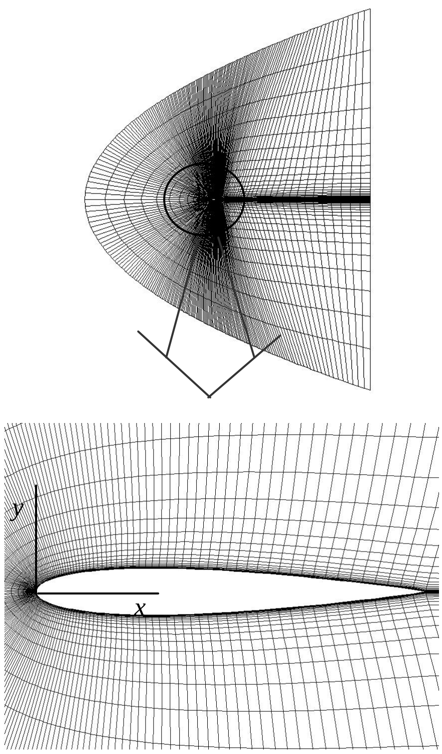


Fig. 1 Computation domain and grid system around a NACA0012 airfoil

The free stream far-field and the wall boundary conditions are applied to the circumferential boundary of the computational domain and the wing surface, respectively. The Mach number based on the free stream velocity and the sound speed is changed from 0.03 to 0.88 during an accelerating flow process.

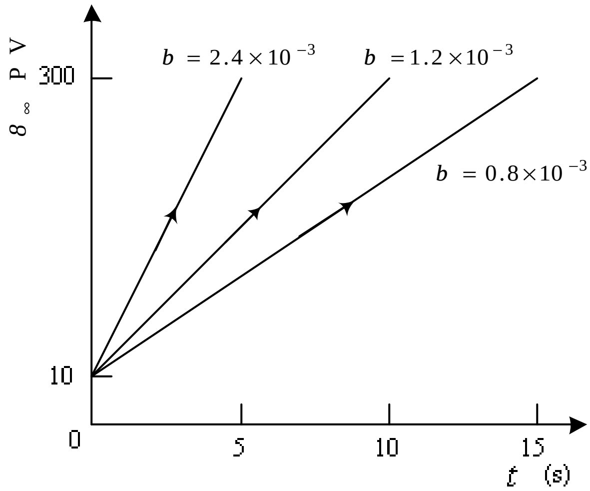


Fig. 2 Three types of accelerating free streams used

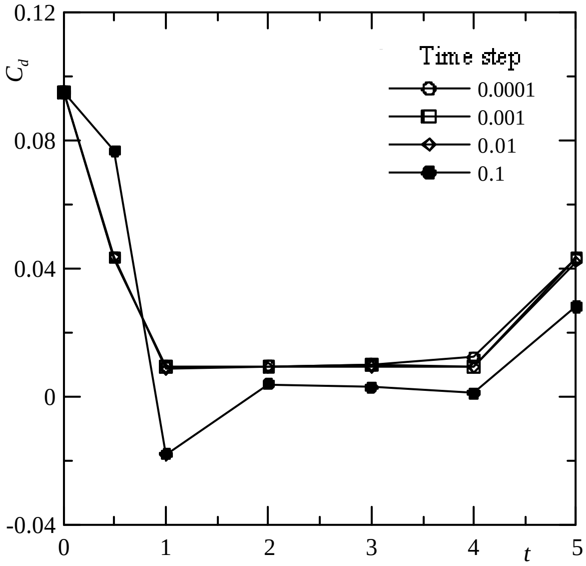
Fig.2 shows the three different types of the accelerating free stream used in this unsteady calculation. The nondimensional acceleration factor b is defined as:

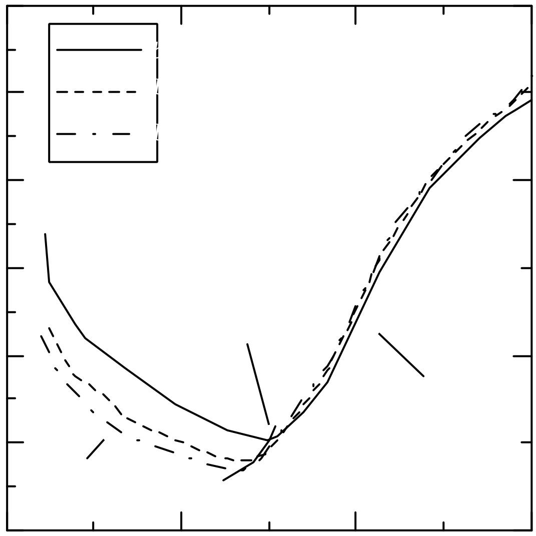
$$b=\frac{(U_f-U_i)/\Delta t}{U_m/T} \tag{4}$$

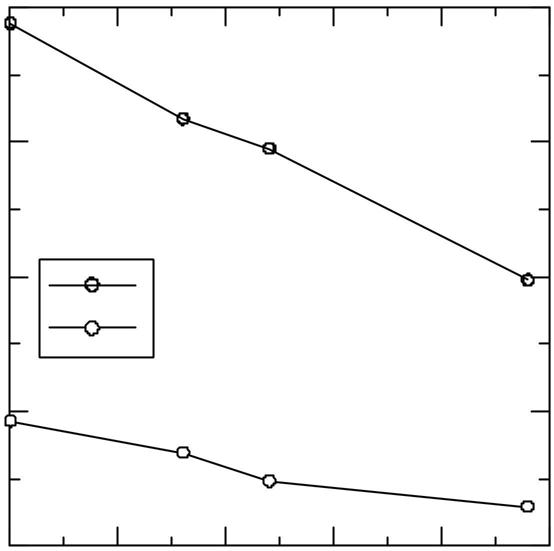
Here, Δt is time length during accelerating from 10m/s to 300m/s, U_f , U_i and U_m are the final, initial and mean speeds of the free steam, respectively, and T is a time that takes when the flow with the initial velocity speed pass the wing. The value of b=0 means a steady calculation.

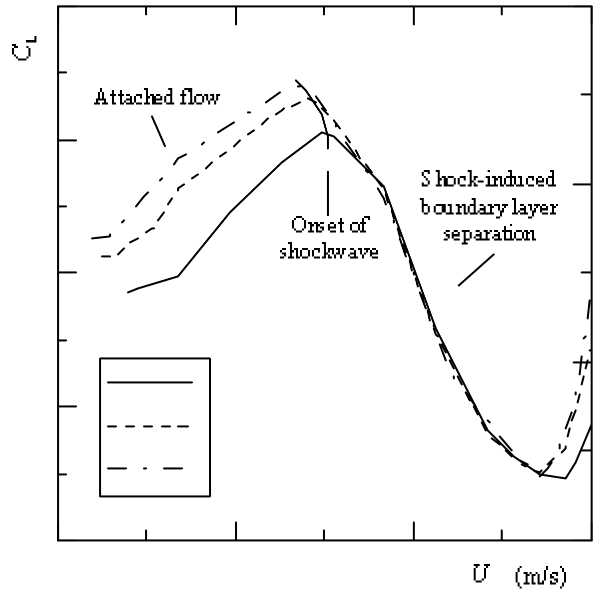
3. REULTS AND DISCUSSION

Fig.3 shows four kinds of the test for determining proper time step of the unsteady calculation at the angle of attack $a=0^o$ and $b=2.4\times10^{-3}$. The time steps used in the figure vary from 10^{-1} to 10^{-4}. As shown in the figure, all the time steps except time step of 10^{-1} have almost same values during the variations of time. To decrease time consumption in the calculation, the time step of 10^{-2} was used.


0.12
0.08
0.04
0
-0.04
C_d
Time step
0.0001
0.001
0.01
0.1
0
1
2
3
4
5
t






C_L
Attached flow
Onset of shockwave
Shock-induced boundary layer separation
U (m/s)

separated from a wing surface. Also, the shockwave moves rearwards and finally reaches near the trailing edge of the aerofoil. The separated region will be decreased.

Fig.6 shows velocity contours with the variations in the free stream velocity at $a = 0^{\circ}$. From the figures, it can be seen that the shockwave thickens the boundary layer on the airfoil. Also, it can be found that, as increasing *b*, the shock appears earlier. Figs.7 are corresponding to Fig.6 at $a = 10^{\circ}$. It can be found that the increase in angle of attack induces the shock boundary layer separation at subsonic speeds. Less effect of a on the shock location is found compared with zero angle of attack(Fig.6) during the variations in *b*.

4. CONCLUSIONS

The numerical computations were carried out to elucidate the unsteady flow characteristics around the wing in gradually accelerating/decelerating free stream speeds. The main conclusions are summarized as follows:

1. The increase in the nondimensional acceleration factor moves the shock location to the leading edge of the wing.
2. The movement of shock location to the leading edge was delayed by increasing the angle of attack.
3. The shock boundary layer separation occurred at the subsonic free stream speeds by increasing the angle of attack.

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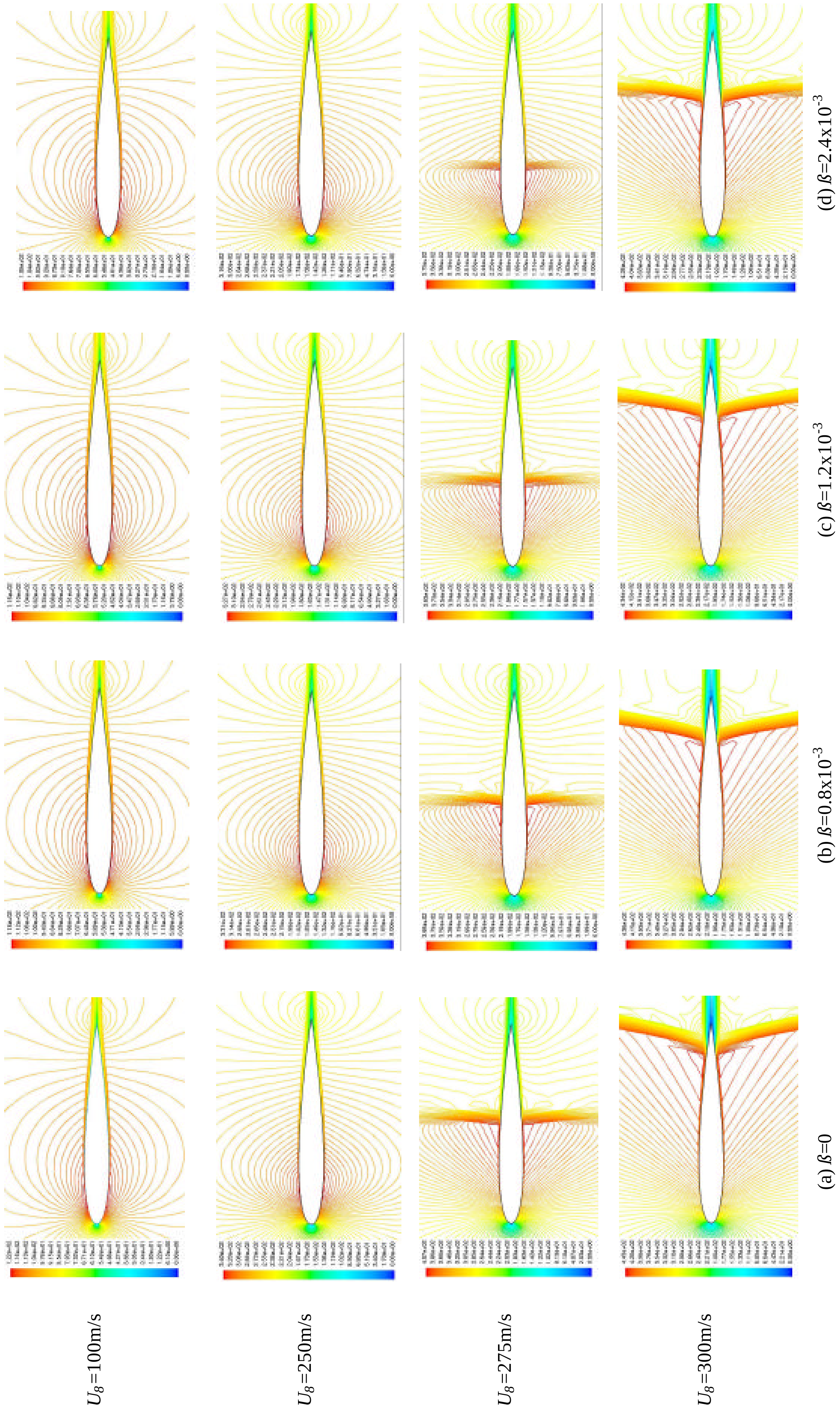

Fig. 6 Velocity contours around the NACA0012 airfoil at α =0 °

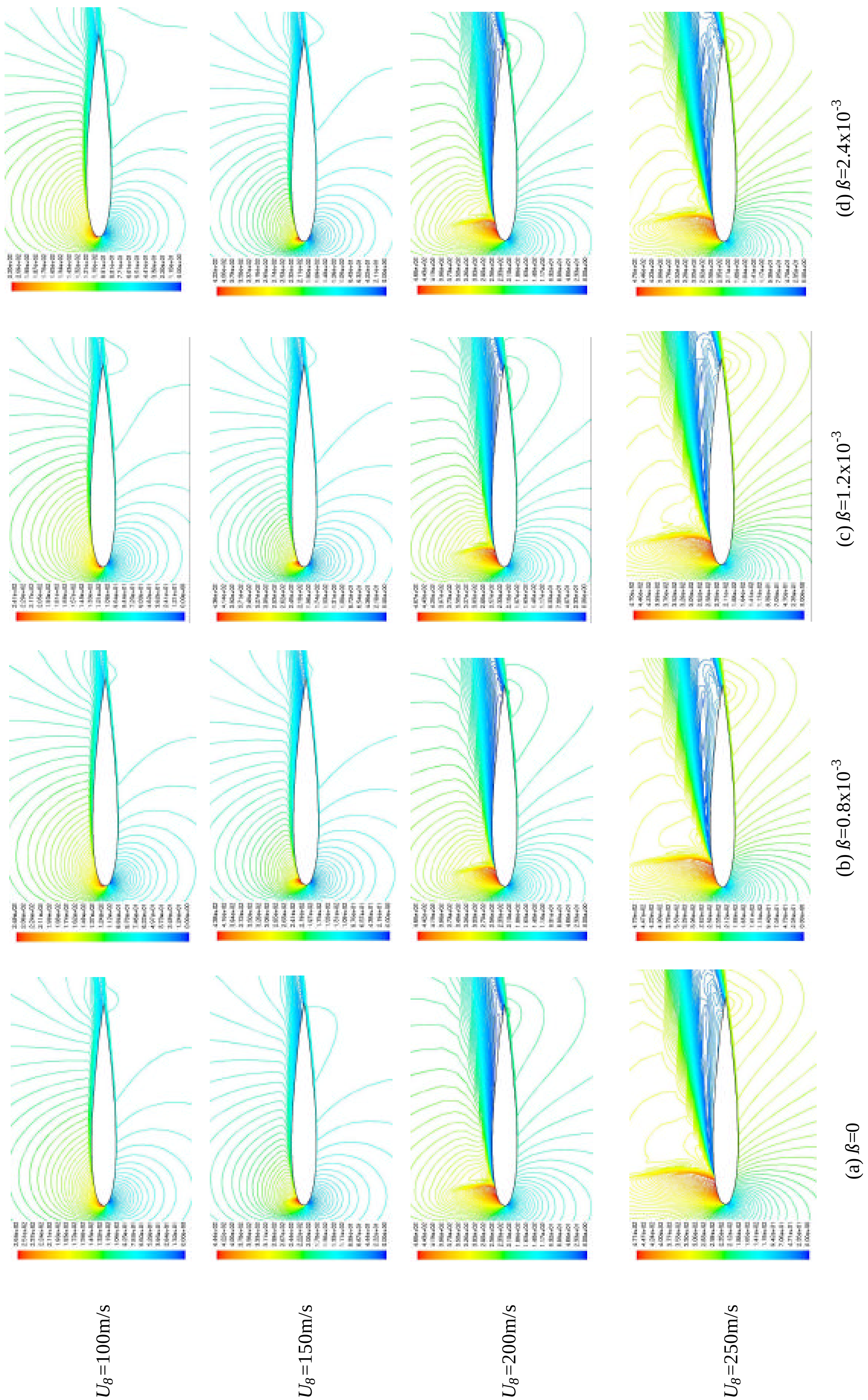

Fig. 7 Velocity contours around the NACA0012 airfoil at α =10°