Available online www.ejaet.com

European Journal of Advances in Engineering and Technology, 2016, 3(7): 47-52

Research Article ISSN: 2394 - 658X

Introducing an Optimized Airfoil Shape Using Panel Method: A Short Report

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ABSTRACT

The present study indicates a direct optimization procedure for finding an airfoil shape which has a relative high lift coefficient when it is compared with the classic airfoil of NACA 0012. Panel Method has been chosen as the methodology to find the pressure coefficient over the upper and lower surface of the airfoil. Analyzing the exerted forces on an airfoil simply results that the effect of boundary layer is negligible on the lift force. This analysis proves the validity of Panel Method as an effective way to obtain the pressure distribution over an airfoil and subsequently the lift coefficient. In this research, we have run an optimization on NACA 0012. By using an algorithm that changes the airfoil shape after each iteration process (this change occurs around the airfoil of NACA 0012) an optimized airfoil shape was achieved. Although, this optimized shape of airfoil possesses a greater lift coefficient, the present work will not be effectively concluded unless other important parameters (including aero- dynamical and manufacturing parameters) of the airfoil are examined.

Key words: Panel Method, Airfoil optimization, NACA 0012, Subsonic aerodynamics

INTRODUCTION

The effectiveness of a floating design is depended on many parameters. These parameters are related to the different fields of aerospace engineering including flight mechanics, aerial structures, propulsion and aerodynamics. Among them, aerodynamics stands as one of the most sensitive factor in the performance of a floating design [1-5]. Coming to this point that aerodynamics deals directly with calculating the lift, drag and other parameters related to the shape of the aerial vehicle, it is worth to seek for the most effective shape which can satisfy the flight conditions of the aerial vehicle. Moreover, since lift and drag forces are directly related to the lift and drag coefficients, even a slight manipulation in aerodynamics of the aerial vehicle may result in significant changes in flight conditions of the vehicle. Airfoil shape has been a classical target to assess the aerodynamic performance of the aerial vehicle so far [3- 9]. Therefore, many studies have been headed to obtain the airfoil shape for desired flight conditions. Considering this that a full simulation of an airfoil requires a full CFD (Computational Fluid Dynamics) simulation, and besides, CFD simulation of an external flow is usually a time consuming route, many attempts have been done in order to simplify the aerodynamic problem of the airfoil [7-14]. Among the proposed methodologies in the literature, it is Panel Method which stands as a fast procedure to calculate the pressure coefficient and subsequently the lift factor of a specific airfoil shape. The idea of Panel Method comes from this fact that the flow over an airfoil (or any other streamline bodies) could be assumed invicid, irrotational and incompressible if the pressure distribution over an airfoil is the only target for subsonic flows [15]. Analysing the forces which are exerted to an airfoil in a fluid media can simply prove the validity of Panel Method for calculating the pressure distribution over an airfoil. In the present work, we have applied the Panel Method for more than 10 billion estimated shapes of airfoil (a direct optimization procedure). This procedure was done for finding the most effective airfoil shape which possesses the greatest lift factor. In the next sections a discussion around the methodology of changing the airfoil shape in each iteration step is provided. Finally, it must be noted that although the acquired airfoil shape during the iteration procedure is the most optimized shape for achieving the greatest lift factor, but further researches are still needed for examining other related parameters to see if this shape is the most effective aerodynamic shape of an airfoil for the low speed flights or not.

A BRIEF INTRODUCTION ON PANEL METHOD

Panel methods are technics for solving incompressible potential flows over 2D or 3D thick bluff bodies. There are several of these Panel Method technics. These methods are applied especially for calculating the pressure distribution over an airfoil. Among them, Vortex Panel Method has the advantage of application for airfoils in different angle of attack. Other Panel Methods whose do not consider the vortex potential flow in their simulation procedure, cannot be applied for asymmetric airfoils or airfoils in different angles of attack. In the present work, Vortex Panel Method is introduced briefly. In 2-D, the airfoil surface is divided into piecewise straight line segments or panels or 'boundary elements' and vortex sheets of strength g are placed on each panel [13]. The philosophy of the existence of lift force on an airfoil comes from this fact that the upper surface boundary layer contains, in general, clockwise rotating vorticity and the lower surface boundary layer contains, in general, counter clockwise vorticity. Because there is more clockwise vorticity than counter clockwise vorticity, there is net clockwise circulation around the airfoil. In panel methods, we replace this boundary layer, which has a small but finite thickness with a thin sheet of vorticity placed just outside the airfoil. This net clockwise circulation around the airfoil can be understood as the existence of lift force on the airfoil. In this model, the vorticity around the airfoil is modelled by assuming vortexes in each panel around the airfoil. At first, the strength of these vortex flows is not identified. So by considering a specific value for the airfoil as the stream line, and using the superposition method for calculating the effect of other panels on a certain panel, the main equation of Vortex Panel Method is obtained. Finally, because we have assumed a certain value for the airfoil as a stream line, this value must be also identified. So, for n numbers of panel, we have n equations and n+ 1 unknown. Here, Kutta condition is applied to balance the equations with the unknowns. Kutta condition states that the pressure above and below the airfoil trailing edge must be equal, and then the flow must smoothly leave the trailing edge in the same direction at the upper and lower edge. Because the methodology of Vortex Panel Method is simply found in the literature of classical methods for solving the potential flow, we have skipped from a detailed discussion around this subject in the present work. Main governing equations of the Vortex Panel Method can be written as:

$$u_\infty y - v_\infty x - \frac{1}{2\pi} \oint \gamma_0 \ln\left(\left|\bar{r} - \bar{r}_0\right|\right) ds_o = C$$

$$\gamma_{Upper} = -\gamma_{lower}$$

In which Eq. 1 is for the cumulative effect of other panel's vortexes on a specific panel and Eq. 2 stands as the Kutta condition [13].

OPTIMIZATION METHODOLOGY AND RESULTS

In this section, we have continued the simulation by Vortex Panel Method for many cases to obtain an airfoil shape which possesses the greatest lift coefficient. A direct optimization procedure was used in which we have assumed 10 panels for the upper surface and 10 for the lower one. These panels were selected on the surface of NACA 0012, because it is already assumed that this airfoil has some identical features in the flight industries. So, we aimed to seek for the best lift coefficient around this airfoil. The selected panels (points) were in an equal interval. Each point was displaced in three different vertical directions during the iteration process. The interval for the vertical displacement of the selected points was about 9mm at first. Therefore, we had 3^{20} numbers of assumed shapes for airfoil. Because the displacement of points was selected to be 9mm, we have automatically ignored the existence of optimized shape in the distances lower than 9mm. So, for being more certain about the optimization process, the former procedure was done for two other displacements of 6mm and 3mm (each new optimization procedure (each new assumed displacement) has been continued on the obtained airfoil shape of the former optimization procedure). Finally, the optimized obtained shape has been smoothened by fitting a correlation on its points (by using non- linear Least Square techniques 2016 [16- 20]). The relations for upper and lower surface of the new airfoil are as:

For the upper surface:

$$y_{upper} = ae^{-((x-b)/c)^2} + de^{-((x-f)/g)^2}$$

In which

a= 0.07166, b= 0.3599, c= 0.4246, d= -0.04015, f= -0.3614, g= 1.032 and e is the Euler's number (e ≈ 2.71828).

For the lower surface:

$$y_{lower} = ae^{-((x-b)/c)^2} + de^{-((x-f)/g)^2}$$

In which a= -0.0136, b= 0.32, c= 0.41, d= 0.0078, f= -0.35, g= 1.03 and e is the Euler's number (e ≈ 2.71828).

So, the new airfoil shape has the most compatibility with NACA 0012 in design but with a considerably greater lift coefficient. The results of the present simulation are provided in Fig. 1 to 26.

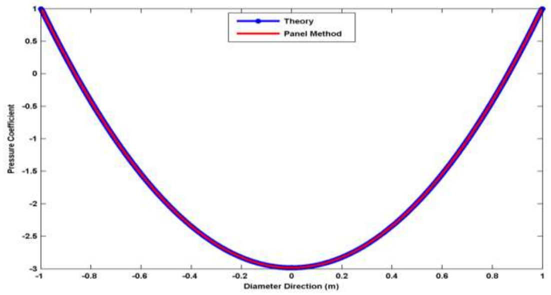

Fig. 1 Pressure coefficient (Cp) around a circle compared with the theoretical solution

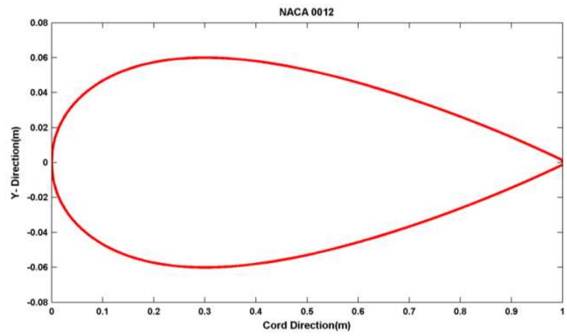

Fig.2 NACA 0012

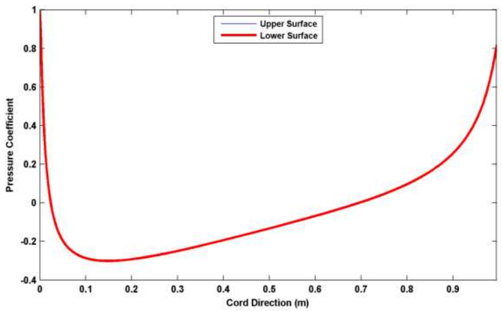

Fig. 3 NACA 0012 in 0 degree angle of attack

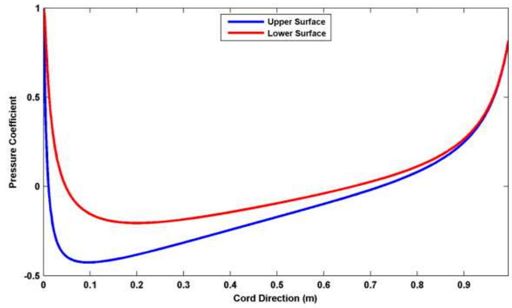

Fig. 4 NACA 0012 in 1 degree angle of attack

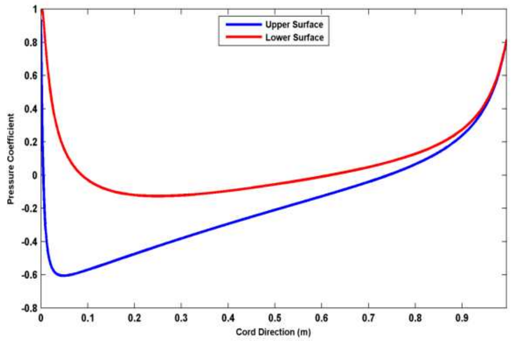

Fig. 5 NACA 0012 in 2 degrees angle of attack

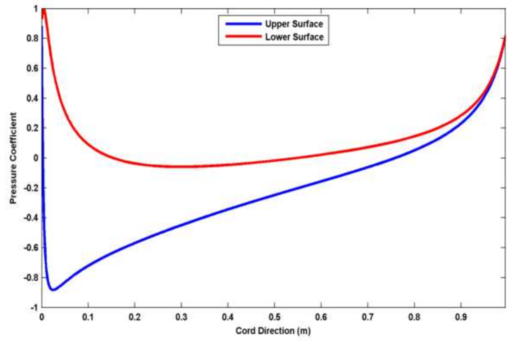

Fig. 6 NACA 0012 in 3 degrees angle of attack

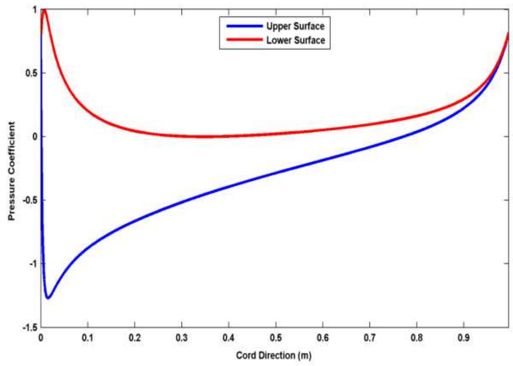

Fig. 7 NACA 0012 in 4 degrees angle of attack

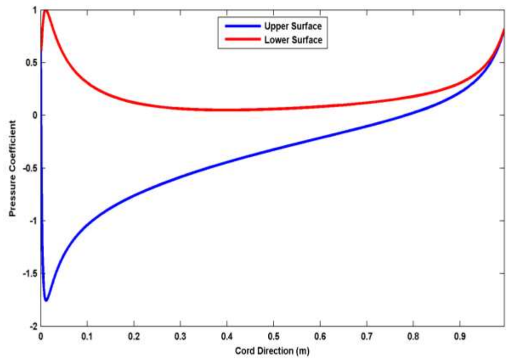

Fig. 8 NACA 0012 in 5 degrees angle of attack

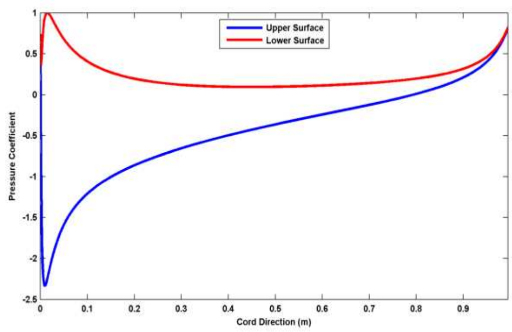

Fig. 9 NACA 0012 in 6 degrees angle of attack

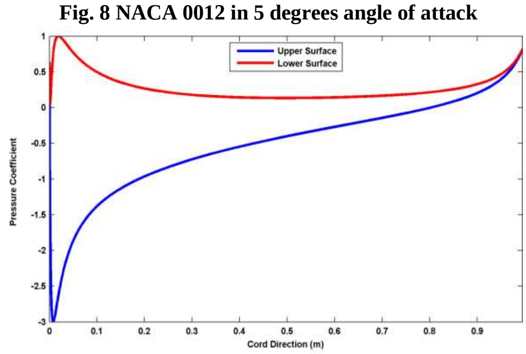

Fig. 10 NACA 0012 in 7 degrees angle of attack

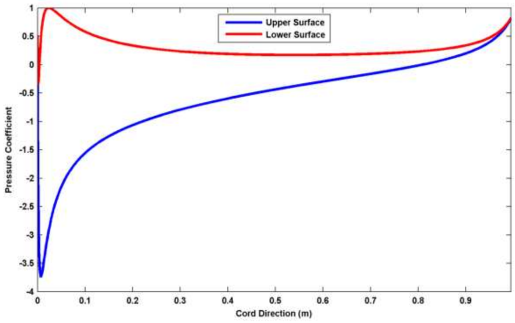

Fig. 11 NACA 0012 in 8 degrees angle of attack

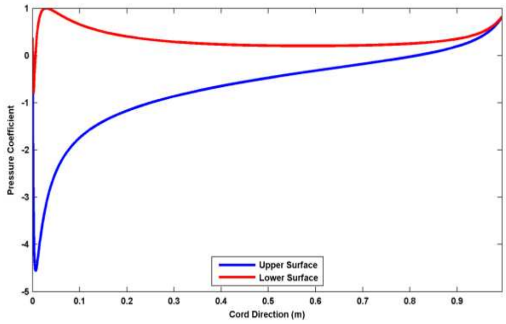

Fig. 12 NACA 0012 in 9 degrees angle of attack

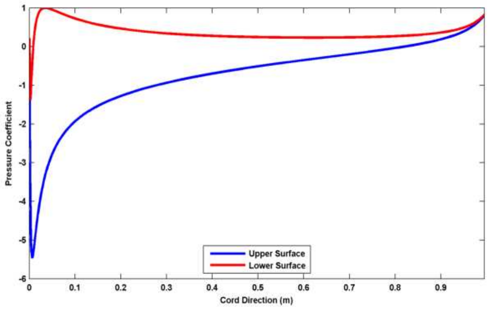

Fig. 13 NACA 0012 in 10 degrees angle of attack

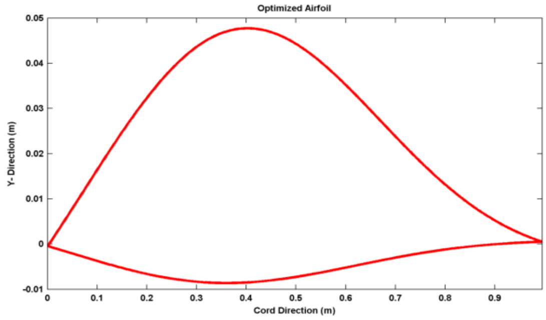

Fig. 14 New Optimized airfoil Shape

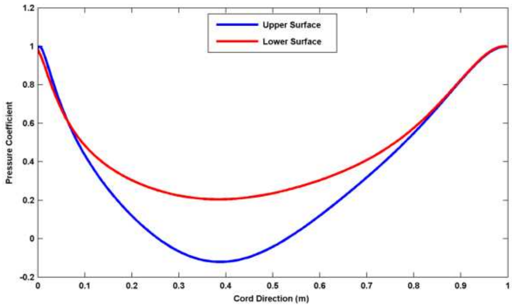

Fig. 15 Optimized airfoil in 0 degree angle of attack

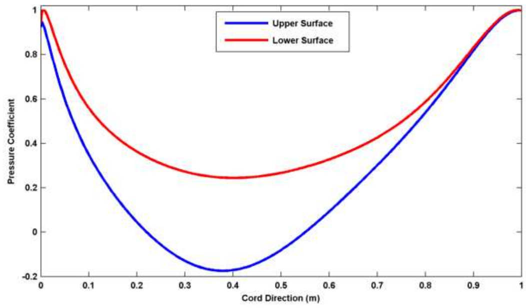

Fig. 16 Optimized airfoil in 1 degree angle of attack

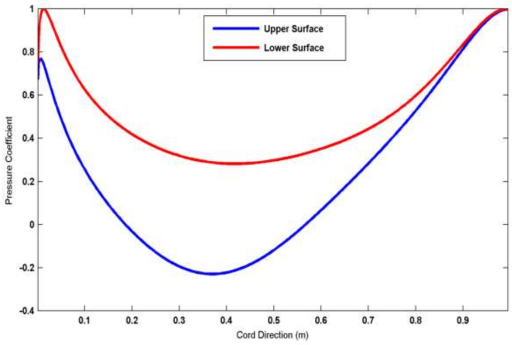

Fig. 17 Optimized airfoil in 2 degrees angle of attack

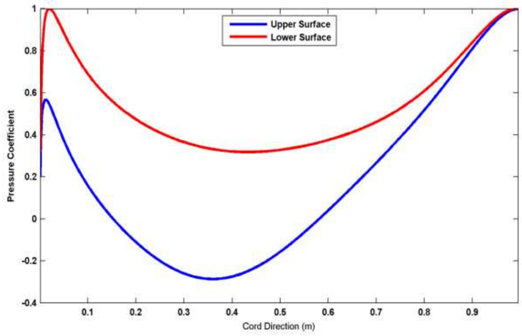

Fig. 18 Optimized airfoil in 3 degrees angle of attack

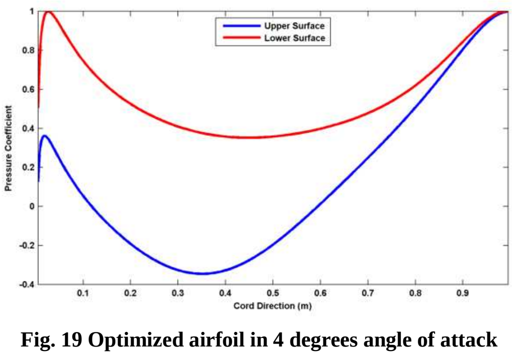

Fig. 19 Optimized airfoil in 4 degrees angle of attack

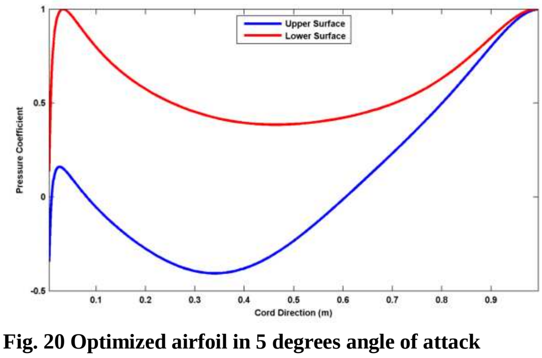

Fig. 20 Optimized airfoil in 5 degrees angle of attack

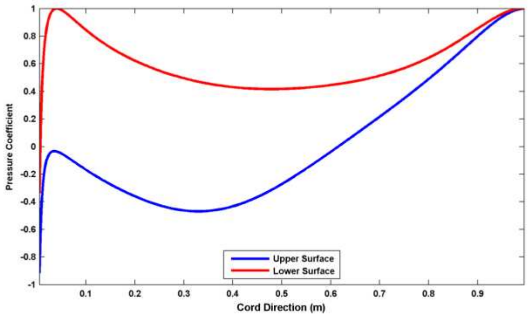

Fig. 21 Optimized airfoil in 6 degrees angle of attack

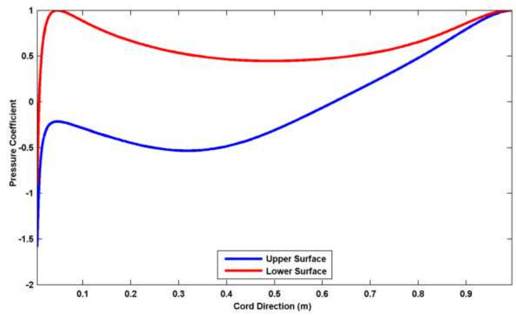

Fig. 22 Optimized airfoil in 7 degrees angle of attack

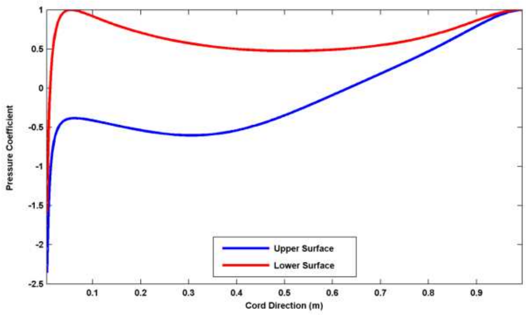

Fig. 23 Optimized airfoil in 8 degrees angle of attack

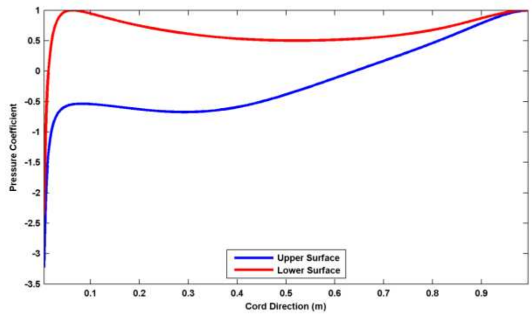

Fig. 24 Optimized airfoil in 9 degrees angle of attack

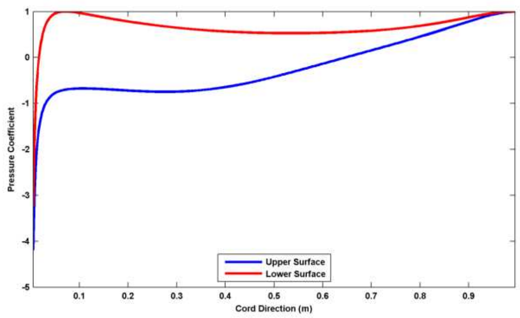

Fig. 25 Optimized airfoil in 10 degrees angle of attack

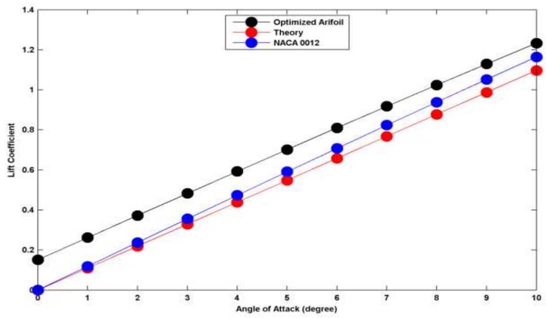

Fig. 26 A comparison of lift coefficient between NACA 0012, optimized airfoil and the theory of thin airfoils and low speed aerodynamics for different angle of attacks

CONCLUSIONS

Vortex Panel Method was used as a fast way to calculate the lift factor. First, we have assumed that NACA 0012 has some design advantages, so by selecting some finite points (panels) on the surface of NACA 0012, the optimizing procedure has begun. We have changed the position of each point in vertical options for creating the new shapes of airfoil. At the end of the optimization process (searching process), specific positions of the selected points have been derived from the developed code. Finally, by fitting Gaussian functions on these points by using Least Square technics 2016 [16- 20], the estimated relation between these points has been achieved. It is worth to state that although the new airfoil has a greater lift coefficient in the comparison with NACA 0012, and also it seems that the new proposed shape is the best shape around NACA 0012 whose possesses the most lift coefficient, but as it was mentioned earlier, further researches are still required to investigate other related parameters (including drag factor, stability factors and so forth) to see if this airfoil has the most effective aerodynamic shape of an airfoil for the low speed flights or not.

Acknowledgment
Authors would like to thank Dr. Mani Fathali for fulfilling the present research.

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