

A Heuristic Approach for Design and Calculation of Pressure Distribution over Naca 4 Digit Airfoil

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Abstract: In recent years of technological perspective, the pressure distribution of airfoil has come to be a wide extent of studies using experimental, theoretical and computational code solving methods. In this hierarchy, it is more predominant to calculate the value of pressure distribution directly from the airfoil without any mode of additional requirement of experimental or computational one. In this paper, we present the method of plotting the NACA 4 digit airfoil and pressure distribution around it at zero degree angle of attack. The obtained results are compared and contrasted with experimental and computational calculations. The empirical formula of C_p is given and calculated for NACA 4 digit airfoil and MATLAB codes.

Keywords: NACA 4 digit, pressure distribution formula, MATLAB codes, etc.

I. Introduction

NACA airfoil profiles are widely used in the aircraft industry for producing lift on the wing span. The geometry of airfoil dictates the airfoil performance and determines its relevance for a specific application. The present analysis deals with the analysis of pressure distribution over the airfoil surfaces along with the lift and moment characteristics. They are commonly expressed in the form of drag polar for the NACA series and represent the aerodynamic performance at various flow field conditions. The Reynolds number is often used to characterize the flow conditions and predict the behavior of airfoils at various operating flow conditions both for the viscous and in viscid fluids. Typically the results from the experiments serve as the simulation tool in the aircraft industry for comparison with actual performance of the wing. Panel methods are modern numerical techniques which are quick to execute and predict results compared to the traditional experimental methods that are cumbersome procedures and time consuming. Apart from Panel methods, there are MATLAB codes contains empirical formula for calculating accurate value of pressure distribution along with lift and moment.

II. Naca Airfoil Geometry:

Abbott and Von Doenhoff were the first derived the NACA four-digit wing sections in 1932. It was found that the thickness distribution of efficient wing sections such as the Gottingen 398 and the Clark Y were nearly the same when the maximum thickness was set equal to the same value.

For NACA four-digit series, the first digit represents the maximum camber in hundredths of chord; the second digit represents position of maximum chamber in tenths of chord and the last two digits represents the maximum thickness in percent of chord.

The thickness distribution for the NACA four-digit sections was selected to correspond closely to that for these earlier wing sections and is given by the following equations:

$$y_t = \pm 5ct [0.2969\sqrt{\epsilon} - 0.1260\epsilon - 0.3516\epsilon^2 + 0.2843\epsilon^3 - 0.1015\epsilon^4]$$

Where,

Where, t – Maximum thickness expressed as fraction of chord.

$\epsilon = x/c$.

The leading edge radius is given by $r_t = 1.1019ct^2$

Thus the ordinate at any point is directly proportional to the thickness ratio and the leading edge radius varies as the square of the thickness ratio.

The equations used to define the camberline are:

$$y_c = \frac{mc}{P^2} (2P\epsilon - \epsilon^2) : \epsilon \leq P$$

$$y_c = \frac{mc}{(1-P)^2} [(1-2P) + 2P\varepsilon - \varepsilon^2] : \varepsilon \geq P$$

To determine the lifting characteristics using thin-aerofoil theory the camber-line slope has to be expressed as a Fourier series. Differentiating the above equation with respect to x gives,

$$\frac{dy_c}{dx} = \frac{d\left(\frac{y_c}{c}\right)}{d\varepsilon} = \frac{2m}{P^2} (P - \varepsilon) : \varepsilon \leq P$$

$$\frac{dy_c}{dx} = \frac{d\left(\frac{y_c}{c}\right)}{d\varepsilon} = \frac{2m}{(1-P)^2} (P - \varepsilon) : \varepsilon \geq P \quad C_p$$

Changing the variables from ε to θ , where $\varepsilon = (1 - \cos\theta)/2$ gives

$$\frac{dy_c}{dx} = \frac{m}{P^2} (2P - 1 + \cos\theta) : \theta \leq \theta_p$$

$$\frac{dy_c}{dx} = \frac{m}{(1-P)^2} (2P - 1 + \cos\theta) : \theta \leq \theta_p$$

Where, θ_p is the value of corresponding to $x = pc$.

To find the pressure distribution over airfoil, using thin airfoil theory the formula for coefficient of pressure is solved as,

$$C_p = -\frac{2}{\pi} \int_0^c \frac{dy_c}{dx_1} \frac{x}{x - x_1} dx_1$$

Integrating the above equation into 4 equal segments, by separating the airfoil in camberwise and chordwise gives finite value of C_p in leading edge and trailing edge of the airfoil. The above formula results thickness problem and gives only the results of accurate inviscous incompressible flow, but in practice the flow regime is viscous incompressible flow, due to viscous boundary layer separation occurs in the thickened section of the airfoil.

III. Design And Calculation Of C_p Using Matlab Code:

3.1. FLOWCHART MATLAB CODE:

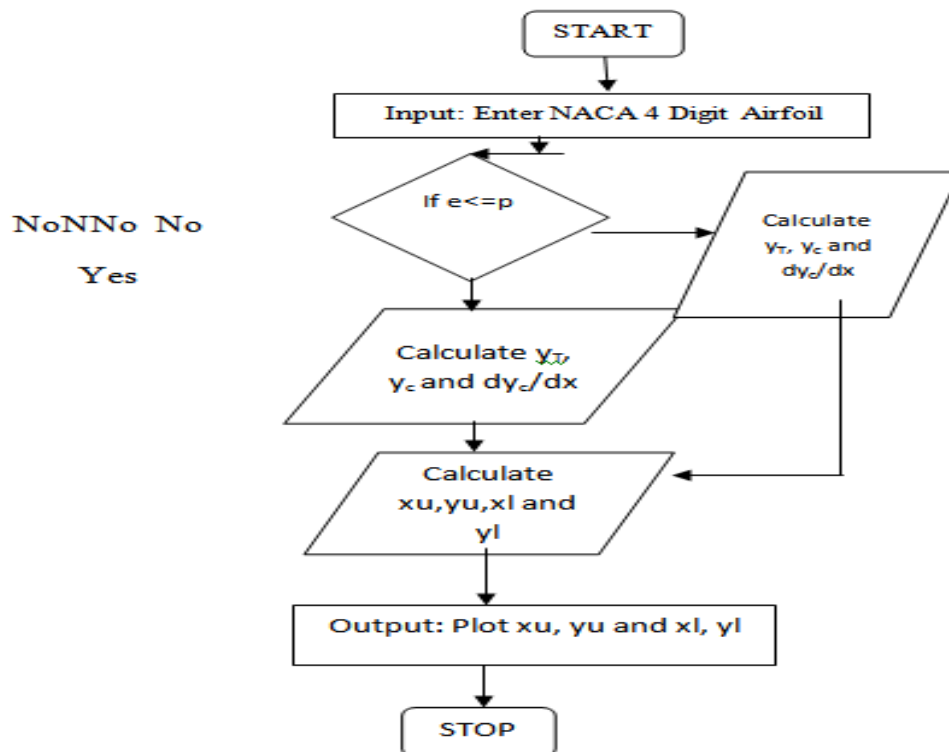


Fig.3.1 Flowchart for design of NACA 4 digit airfoil using MATLAB code.

Use str2double command for all the input numerical in real and imaginary part, the ratio of them is 1:1. The value of ϵ is taken as 0 to 1 as well as conditions statements like for, if and else if command was purposefully encoded in the MATLAB code. These codes generate a nested loop for camber line thickness and maximum thickness of the camber of the airfoil. The result obtained are x_u, y_u and x_l, y_l are in nested columns. The results are plotted using Plot command. The NACA 4 digit airfoil of any type of numerical can easily be plotted using the above MATLAB code.

3.2 FLOWCHART FOR PRESSURE DISTRIBUTION CALCULATION:

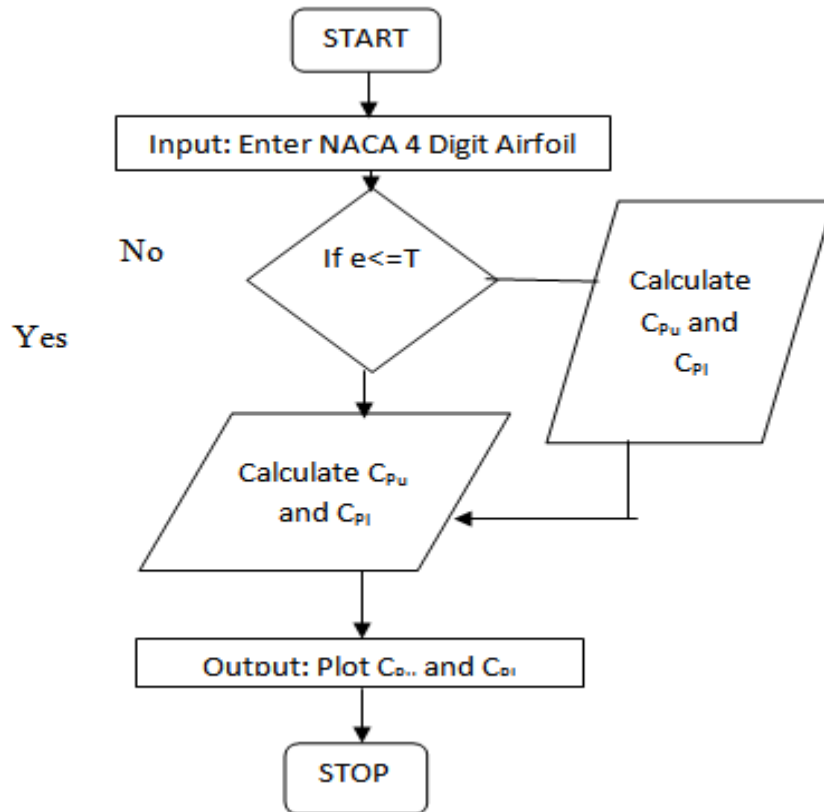


Fig.3.2. Flowchart for calculating the pressure distribution around the airfoil using MATLAB code.

This is the empirical formula (using thin airfoil theory), the integral equation of C_p was taken here for calculating the pressure distribution for any NACA 4 digit airfoil by knowing its airfoil configuration and the value of C_p is calculated and plotted. The upper surface shows negative C_p value and the lower surface shows positive to negative variation of C_p value for various x/c distance ratio.

3.3. GEOMETRY OF NACA 4414:

Utilizing the above MATLAB code, generate the airfoil as shown,

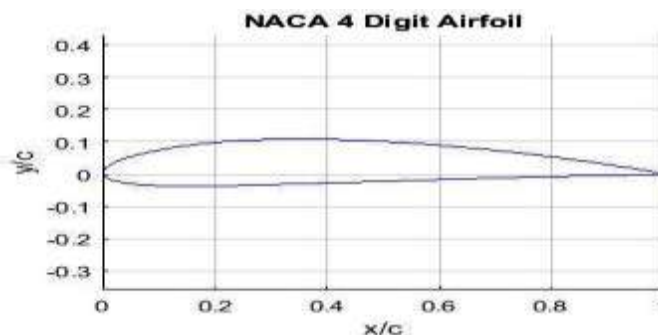


Fig.3.3. NACA 4414 Geometry.

3.4. C_p OF NACA 4414 USING MATLAB CODE:

Utilizing the above MATLAB code, it was easy to generate C_p for any NACA airfoil based on thin airfoil theory application. We can generate the value of C_p in front upper camber and rear upper camber as well as in front lower camber and rear lower camber using the empirical relation are plotted and shown in the Fig.3.4. The only disadvantage of olden technique shows infinite values starting from leading edge to trailing edge of the airfoil and heuristic thickness problem but here in this calculation finite results are obtained with values of C_p minimum extends to infinite slip and value of C_p maximum is always found to be less than or equal to 1. This code is futuristically developed and it is helpful for pre analysis part in selection of airfoil in various design constraint parameters.

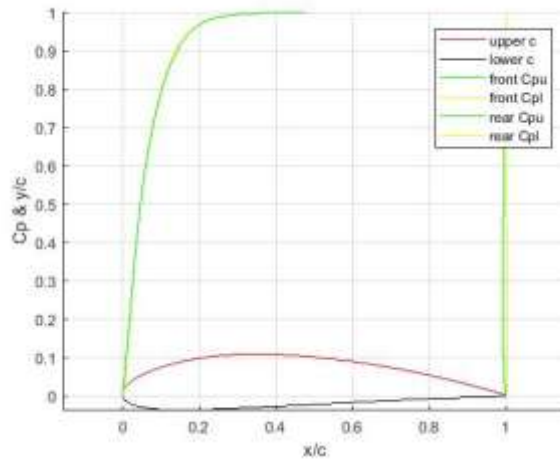


Fig.3.4. C_p value for NACA 4414 airfoil

3.5. COMPARISON OF CALCULATED VALUE OF MATLAB CODE C_p WITH EXPERIMENTAL VALUE:

S.No.	x/c	EXPERIMENTAL C_p VALUE	MATLAB CODE C_p VALUE
1	0.05	0.57	0.34
2	0.17	0.50	0.73
3	0.30	0.00	0.90
4	0.46	0.71	0.90
5	0.67	0.20	1.00
6	0.85	0.57	0.94
7	0.86	0.50	0.93
8	0.68	0.00	1.00
9	0.48	0.00	0.95
10	0.30	0.25	0.90
11	0.17	0.20	0.73
12	0.05	-0.33	0.34

Table 3.1. Experimental C_p vs MCODE

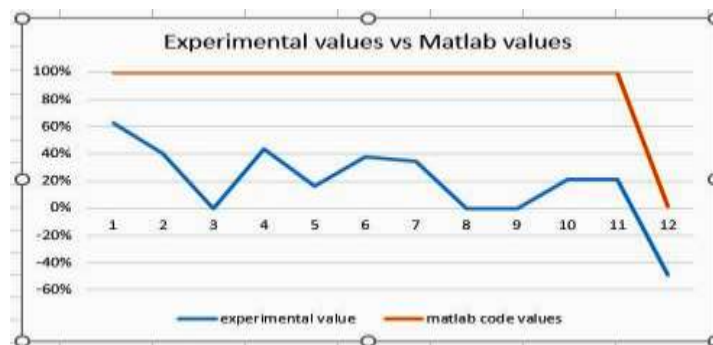


Fig.3.5. Comparison of experimental value vs MATLAB code value of C_p .

IV. CALCULATION OF C_L AND $C_{Mc/4}$:

Based on the application of thin airfoil theory, the calculation of coefficient of lift and moment coefficient at the quarter chord for any type of 4 digit airfoil can be easily undergone utilizing the MATLAB code to find the better result of zero degree angle of attack for selection of airfoil in design parameters. The Fourier's series coefficients like A_0 , A_1 and A_2 can be easily estimated for NACA 4 digit airfoil using the MATLAB code and it was very easy to estimate the value of C_L and $C_{Mc/4}$.

$$C_L = \pi(A_1 - 2A_0)$$
$$C_{Mc/4} = -\frac{\pi}{4}(A_1 - A_2)$$

Where, A_0 , A_1 and A_2 are Fourier's coefficients.

The values obtained from MATLAB code are as follows:

$$A_0=0.0090$$

$$A_1=0.3355$$

$$A_2=0.0148$$

$$C_L=0.9974$$

$$C_{Mc/4}=-0.2518$$

V. Future Scope Of This Work

This work carries out an important feature of estimation of pressure distribution, lift and moment characteristics with theoretical approach helpful for the engineers as well as design analyst to develop the selection of NACA 4 digit airfoil with high accuracy.

The future work is to determine the variation of C_p and C_L for various angle of attack studies in the mode of optimization technique. There are various optimization tools in MATLAB code, whereas tools like GA and MOGA are utilized and many where approaching the new type of airfoil design and analysis framework. For the same optimized airfoil using this work, it is easy to analyze with the help of this MATLAB code.

VI. Conclusion

The results of NACA 4414 airfoil and its flow calibrations are calculated in both experimental approach as well as MATLAB code (Theoretical Approach). The results are compared and now it is easy to find the value of C_p distribution directly from the MATLAB code for any NACA 4 digit airfoil and it is concluded to have the better results from the experimental one due to boundary separation of viscous flow effect.

References

- [1]. H. Abbot, A von Doenhoff., "Theory of Wing Sections, Including summary of airfoil data", Dover publications Inc. New York.
- [2]. Haughton, Carpenter, "Aerodynamics for Engineering students", 6th Edition, Elsevier publications.
- [3]. Vasishtabargava, ChKoteswara Rao, Y.D. Dwivedi, "Pressure Distribution & Aerodynamic characteristics of NACA Airfoils using Computational Panel Method for 2D Lifting flow", International Journal of Engineering Research ISSN:2319-6890(online),2347-5013(print)Volume No.6, Issue No.1, pp :18-23, Jan.2017.
- [4]. Charles L Ladson, "Computer Program to obtain ordinates for NACA airfoils", Langley Research Center. Hampton, Virginia.
- [5]. Braslow, A. L. and Knox, E. C. (1958), "Simplified method for determination of critical height distributed roughness particles for boundary-layer transition at mach numbers from 0 to 5. NACA TN 4363."
- [6]. http://m-lig.ae.illinois.edu/ads/coord_database.html?
- [7]. <http://www.mh-aerotoools.de/airfoils/velocitydistribution.html>.