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A Nonconventional Approach of Designing Airfoils

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Abstract

An airfoil is a cross section of airplanes wings. It is very much difficult for a researcher to perform his test on the whole aircraft wing. So a test using small cross section of wings or airfoils is performed by researchers. For experimental purposes airfoils need to be constructed. For constructing a real airfoil model an airfoil profile is needed. For obtaining airfoil profiles coordinates are needed. There are many available online sources for these coordinates. But this study will show how someone can obtain required coordinates for drawing airfoil profile without taking any information from online available sources. For obtaining airfoil co ordinates generalized equations for airfoils are used. The whole coordinate generation is performed using programming languages. In this case C programming is used. With the help of these coordinates generated by c programming it is possible to draw airfoil profiles hence an airfoil profile for NACA-4415 airfoil is also shown. The whole study is based on NACA four digit airfoils. In the same fashion study based on NACA five digit and NACA six digit can be performed.

Keywords: Airfoil, Drag, Equation, Source Code, Wing;

1. Introduction

An airplane is a transportation media with wings having one or more engines that enable it to fly through the air .These wings of airplane creating pressure difference between upper and bottom surfaces which in turn producing lift and helps the airplane to fly [1]. Airfoils are the cross section of these wings. Design of airfoil varies from airplane to airplane [2] .Researchers trying for centuries for classifying these various types of airfoils. Among those U.S. National Advisory Committee for Aeronautics or NACA is more successful. NACA classified airfoils in different groups such as four digit series, five digit series etc. Each series of airfoils has their unique characteristics. Each series of airfoils has their unique design features. Upon growing popularities of NACA airfoils researchers are interested on these airfoils. A lot of investigations are performed in different countries around the world. But most of the investigations used online available coordinates for design or drawing airfoil profiles. This study try to make an exception by doing the design or drawing of airfoils using self originated coordinates using programming languages.

2. Equations for Generating Coordinates

Profiles of all NACA airfoils obtained by combining a mean camber line and a thickness distribution [4]. In order to obtain these mean camber line and thickness distributions coordinates are needed. There are generalized equations for all NACA airfoil series. In the present study equations for NACA four digit series are utilized.

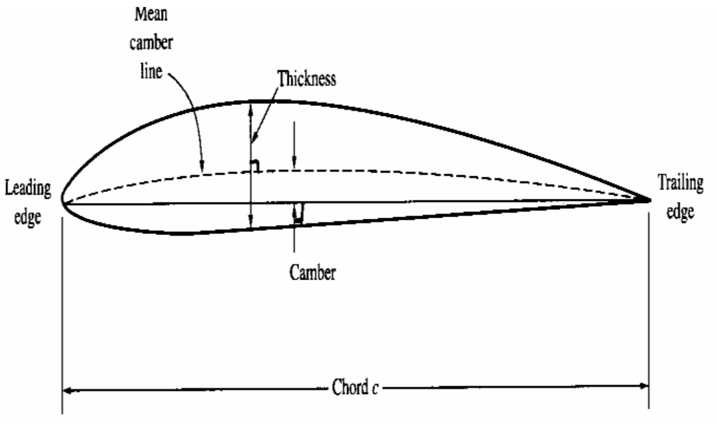


Fig. 2.1. A typical airfoil

The equations for coordinates above and below the line extending along the length of the airfoil is given below this can be used for determining both thickness coordinates and camber coordinates

$$y_c = \frac{n}{r^2}(2rx - x^2) \quad \text{from } x = 0 \text{ to } x = r \tag{1}$$

$$y_c = \frac{n}{(1-r)^2}[(1-2r) + 2rx - x^2] \quad \text{from } x=r \text{ to } x=c \tag{2}$$

Where,
x = coordinates along the chord of the airfoil.
n = highest camber in tenths of the chord.
r =position of the highest camber along the length in tenths of chord.
The thickness distribution for both direction of airfoil surface can be calculated from the following equations providing the value of q.

$$\pm y_t = \frac{q}{0.2}\left(.2969\sqrt{x} - 0.1260x - 0.351x^2 + 0.2843x^3 - 0.1015x^4\right) \tag{3}$$

Where,

q= highest airfoil thickness in tenths of chord

After obtaining the above value and entering the values in the following equations final coordinates of upper surface (X_U,Y_U) and lower surface (X_L, Y_L) can be obtained.

$$x_u = x - y_t \sin\theta \; y_u = y_c + y_t \cos\theta$$
$$x_L = x + y_t \sin\theta \; y_L = y_c - y_t \cos\theta$$

Where θ = arc $\tan\frac{dy_c}{dx}$

As θ is very small

$\sin\theta = 0, \cos\theta = 1$

Hence, X_U= xY_U= y_c+ y_t
X_L= x Y_L= y_c - y_t

The above mentioned equations can be used for calculating the necessary coordinates for obtaining desired airfoil equations. These are generalized equations for four digit series. With the help of these equations a generalized source code is designed.

3. Source Code for Generating Coordinates

The Using the generalized equations for four digit series a generalized source code is designed. This source code can be written in any programming languages and can be used for all four digit series airfoils. Here the c compiler is choosen.

```
#include<stdio.h>
#include<conio.h>
#include<math.h>
void main()
{
    float yt[2000],yc[2000],yu[2000],yl[2000],xu[2000],xl[2000],t,x[2000],m,p,c,n;
    int f,k;
    printf("please enter the chord length:");
    scanf("%f",&c);
    printf("\n please enter the maximum camber:");
    scanf("%f",&m);
    printf("\n please enter the position of camber from leading edge:");
    scanf("%f",&p);
    printf("\n please enter maximum thickness:");
    scanf("%f",&t);
    printf("How many coordinates you want to generate for drawing profile \n note that:the more coordinates the smoother the profile is?");
    scanf("%f",&n);
    printf("The obtained Coordinate of mean camber line are While thickness is:");
    for(f=0;f<=n;f++)
    {x[0]=0;
      x[f+1]=x[f]+c/n;}
      for(f=1;f<=n;f++)
    {
printf("\nx[%d]=%f",f,x[f]) ;
    if(x[f]<=p)
   {

    yc[f]=(m/(p*p))*(2*p*x[f]-x[f]*x[f]);}
    if(x[f]>p)
    {
       yc[f]=(m/((1-p)*(1-p)))*(1-2*p+2*p*x[f]-x[f]*x[f]);
       }
    printf("\tyc[%d]=%f",f,yc[f]);
    yt[f]=5*t*(.2969*(sqrt(x[f]))-.1260*x[f]-0.3516*x[f]*x[f]+0.2843*x[f]*x[f]*x[f]-
0.1015*x[f]*x[f]*x[f]*x[f]);
   printf("\t\tyt[%d]=%f",f,yt[f]);

    xu[f]=x[f];
    yu[f]=yc[f]+yt[f];
    xl[f]=x[f];
    yl[f]=yc[f]-yt[f];}
    printf("\n The obtained upper surface coordinates are:");
    for(f=1;f<=n;f++)
    {printf("\nxu[%d]=%f   yu[%d]=%f",f,xu[f],f,yu[f]);}


    printf("\n The lower surface coordinates are:");
     for(f=1;f<=n;f++)
   {
    printf("\nxl[%d]=%f   yl[%d]=%f",f,xl[f],f,yl[f]);}
```

}

The advantage of using this program is that available sources provide only a few coordinates. But using this program someone can have more coordinates than available sources. Which will results in a better airfoil profile .The more points the more smooth profile.

After running the program in code block following windows will appear:

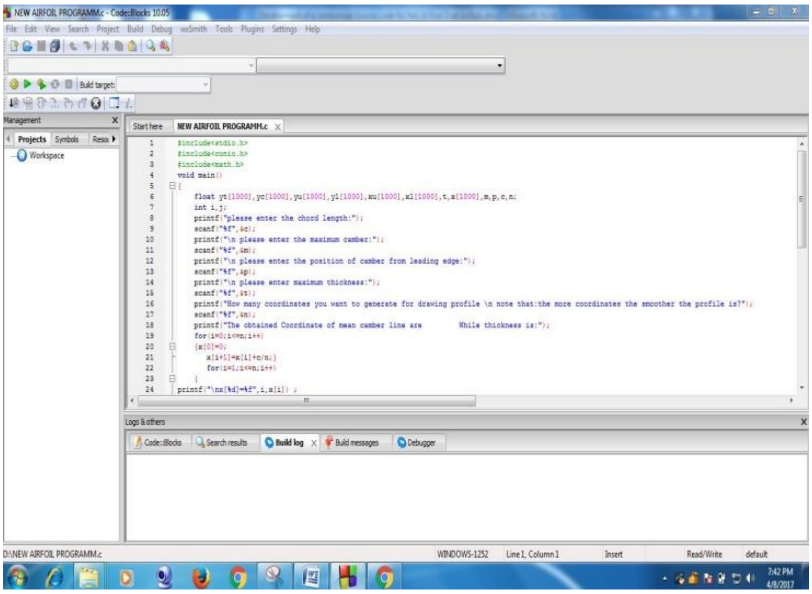


Fig 3.1.Picture of windows that will appear after inserting the program in the code block compiler

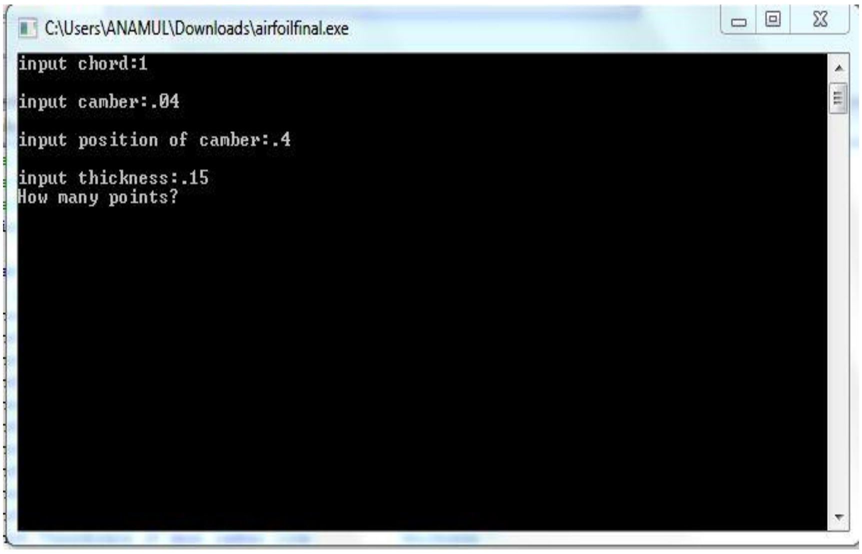
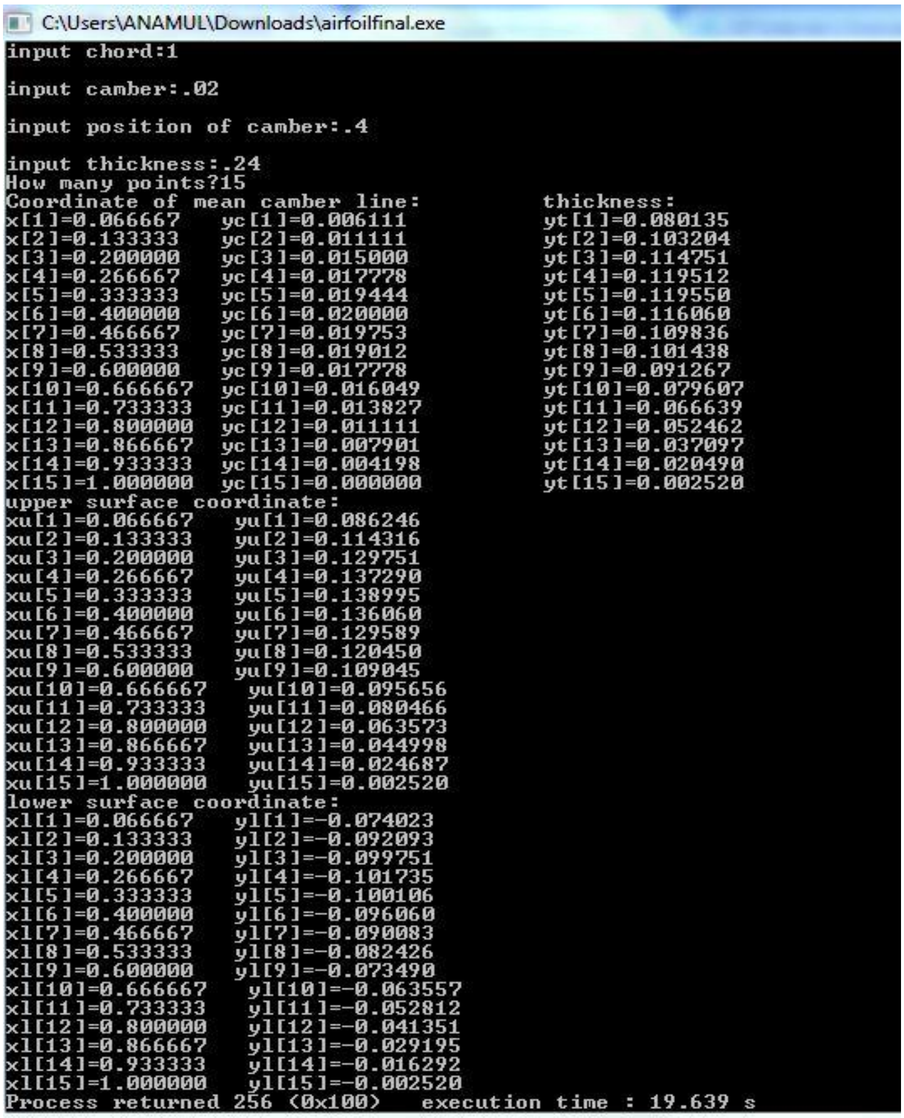


Fig 3.2.Picture of windows that will appear after running the program in the code block compiler



```
C:\Users\ANAMUL\Downloads\airfoilfinal.exe
input chord:1

input camber:.02

input position of camber:.4

input thickness:.24
How many points?15
Coordinate of mean camber line:               thickness:
x[1]=0.066667   yc[1]=0.006111                yt[1]=0.080135
x[2]=0.133333   yc[2]=0.011111                yt[2]=0.103204
x[3]=0.200000   yc[3]=0.015000                yt[3]=0.114751
x[4]=0.266667   yc[4]=0.017778                yt[4]=0.119512
x[5]=0.333333   yc[5]=0.019444                yt[5]=0.119550
x[6]=0.400000   yc[6]=0.020000                yt[6]=0.116060
x[7]=0.466667   yc[7]=0.019753                yt[7]=0.109836
x[8]=0.533333   yc[8]=0.019012                yt[8]=0.101438
x[9]=0.600000   yc[9]=0.017778                yt[9]=0.091267
x[10]=0.666667  yc[10]=0.016049               yt[10]=0.079607
x[11]=0.733333  yc[11]=0.013827               yt[11]=0.066639
x[12]=0.800000  yc[12]=0.011111               yt[12]=0.052462
x[13]=0.866667  yc[13]=0.007901               yt[13]=0.037097
x[14]=0.933333  yc[14]=0.004198               yt[14]=0.020490
x[15]=1.000000  yc[15]=0.000000               yt[15]=0.002520
upper surface coordinate:
xu[1]=0.066667   yu[1]=0.086246
xu[2]=0.133333   yu[2]=0.114316
xu[3]=0.200000   yu[3]=0.129751
xu[4]=0.266667   yu[4]=0.137290
xu[5]=0.333333   yu[5]=0.138995
xu[6]=0.400000   yu[6]=0.136060
xu[7]=0.466667   yu[7]=0.129589
xu[8]=0.533333   yu[8]=0.120450
xu[9]=0.600000   yu[9]=0.109045
xu[10]=0.666667   yu[10]=0.095656
xu[11]=0.733333   yu[11]=0.080466
xu[12]=0.800000   yu[12]=0.063573
xu[13]=0.866667   yu[13]=0.044998
xu[14]=0.933333   yu[14]=0.024687
xu[15]=1.000000   yu[15]=0.002520
lower surface coordinate:
xl[1]=0.066667   yl[1]=-0.074023
xl[2]=0.133333   yl[2]=-0.092093
xl[3]=0.200000   yl[3]=-0.099751
xl[4]=0.266667   yl[4]=-0.101735
xl[5]=0.333333   yl[5]=-0.100106
xl[6]=0.400000   yl[6]=-0.096060
xl[7]=0.466667   yl[7]=-0.090083
xl[8]=0.533333   yl[8]=-0.082426
xl[9]=0.600000   yl[9]=-0.073490
xl[10]=0.666667   yl[10]=-0.063557
xl[11]=0.733333   yl[11]=-0.052812
xl[12]=0.800000   yl[12]=-0.041351
xl[13]=0.866667   yl[13]=-0.029195
xl[14]=0.933333   yl[14]=-0.016292
xl[15]=1.000000   yl[15]=-0.002520
Process returned 256 (0x100)   execution time : 19.639 s
```

Fig 3.3.Picture of windows containing results that will appear after testing the program in the code block compiler

4. Data tables for Generating airfoil profile

Below some results of the above coding is shown in tabular form which will further used for generating airfoil profiles of different cross sections

Table-1 : Table of mean camber and thickness distribution coordinates for NACA-4415 airfoil.

X/C	Coordinates of mean camber line	Thickness of airfoil
x[1]=0.066667	yc[1]=0.012222	yt[1]=0.050084
x[2]=0.133333	yc[2]=0.022222	yt[2]=0.064503
x[3]=0.200000	yc[3]=0.030000	yt[3]=0.071719
x[4]=0.266667	yc[4]=0.035556	yt[4]=0.074695
x[5]=0.333333	yc[5]=0.038889	yt[5]=0.074719
x[6]=0.400000	yc[6]=0.040000	yt[6]=0.072538
x[7]=0.466667	yc[7]=0.039506	yt[7]=0.068648
x[8]=0.533333	yc[8]=0.038025	yt[8]=0.063399
x[9]=0.600000	yc[9]=0.035556	yt[9]=0.057042
x[10]=0.666667	yc[10]=0.032099	yt[10]=0.049754

x[11]=0.733333	yc[11]=0.027654	yt[11]=0.041649
x[12]=0.800000	yc[12]=0.022222	yt[12]=0.032789
x[13]=0.866667	yc[13]=0.015802	yt[13]=0.023185
x[14]=0.933333	yc[14]=0.008395	yt[14]=0.012806
x[15]=1.000000	yc[15]=0.000000	yt[15]=0.001575

Table-2 : Table of upper surface coordinates for NACA-4415 airfoil.

X/C	Upper Surface coordinates
xu[1]=0.066667	yu[1]=0.062306
xu[2]=0.133333	yu[2]=0.086725
xu[3]=0.200000	yu[3]=0.101719
xu[4]=0.266667	yu[4]=0.110251
xu[5]=0.333333	yu[5]=0.113608
xu[6]=0.400000	yu[6]=0.112538
xu[7]=0.466667	yu[7]=0.108154
xu[8]=0.533333	yu[8]=0.101423
xu[9]=0.600000	yu[9]=0.092598
xu[10]=0.666667	yu[10]=0.081853
xu[11]=0.733333	yu[11]=0.069304
xu[12]=0.800000	yu[12]=0.055011
xu[13]=0.866667	yu[13]=0.038988
xu[14]=0.933333	yu[14]=0.021201
xu[15]=1.000000	yu[15]=0.001575

Table-3 : Table of lower surface coordinates for NACA-4415 airfoil.

X/C	Lower Surface Coordinates
xl[1]=0.066667	yl[1]=-0.037862
xl[2]=0.133333	yl[2]=-0.042281
xl[3]=0.200000	yl[3]=-0.041719
xl[4]=0.266667	yl[4]=-0.039140
xl[5]=0.333333	yl[5]=-0.035830
xl[6]=0.400000	yl[6]=-0.032538
xl[7]=0.466667	yl[7]=-0.029141
xl[8]=0.533333	yl[8]=-0.025374
xl[9]=0.600000	yl[9]=-0.021487
xl[10]=0.666667	yl[10]=-0.017655
xl[11]=0.733333	yl[11]=-0.013995
xl[12]=0.800000	yl[12]=-0.010567
xl[13]=0.866667	yl[13]=-0.007383
xl[14]=0.933333	yl[14]=-0.004411
xl[15]=1.000000	yl[15]=-0.001575

Table-4 : Table of mean camber and thickness distribution coordinates for NACA-2412 airfoil.

X/C	Coordinates of mean camber line	Thickness of airfoil
x[1]=0.06666	yc[1]=0.00611	yt[1]=0.040067
x[2]=0.13333	yc[2]=0.01111	yt[2]=0.051602
x[3]=0.20000	yc[3]=0.01500	yt[3]=0.057375
x[4]=0.26666	yc[4]=0.01777	yt[4]=0.059756
x[5]=0.33333	yc[5]=0.01944	yt[5]=0.059775
x[6]=0.40000	yc[6]=0.02000	yt[6]=0.058030

x[7]=0.46666	yc[7]=0.01975	yt[7]=0.054918
x[8]=0.53333	yc[8]=0.01901	yt[8]=0.050719
x[9]=0.60000	yc[9]=0.01777	yt[9]=0.045634
x[10]=0.6666	yc[10]=0.0160	yt[10]=0.039803
x[11]=0.7333	yc[11]=0.0138	yt[11]=0.033319
x[12]=0.8000	yc[12]=0.0111	yt[12]=0.026231
x[13]=0.8666	yc[13]=0.0079	yt[13]=0.018548
x[14]=0.9333	yc[14]=0.0041	yt[14]=0.010245
x[15]=1.0000	yc[15]=0.0000	yt[15]=0.001260

Table-5 : Table of upper surface coordinates for NACA-2412 airfoil.

X/C	Upper Surface coordinates
xu[1]=0.066667	yu[1]=0.046178
xu[2]=0.133333	yu[2]=0.062713
xu[3]=0.200000	yu[3]=0.072375
xu[4]=0.266667	yu[4]=0.077534
xu[5]=0.333333	yu[5]=0.079220
xu[6]=0.400000	yu[6]=0.078030
xu[7]=0.466667	yu[7]=0.074671
xu[8]=0.533333	yu[8]=0.069731
xu[9]=0.600000	yu[9]=0.063411
xu[10]=0.666667	yu[10]=0.055853
xu[11]=0.733333	yu[11]=0.047147
xu[12]=0.800000	yu[12]=0.037342
xu[13]=0.866667	yu[13]=0.026450
xu[14]=0.933333	yu[14]=0.014442
xu[15]=1.000000	yu[15]=0.001260

Table-6 : Table of lower surface coordinates for NACA-2412 airfoil.

X/C	Lower Surface Coordinates
xl[1]=0.066667	yl[1]=-0.033956
xl[2]=0.133333	yl[2]=-0.040491
xl[3]=0.200000	yl[3]=-0.042375
xl[4]=0.266667	yl[4]=-0.041978
xl[5]=0.333333	yl[5]=-0.040331
xl[6]=0.400000	yl[6]=-0.038030
xl[7]=0.466667	yl[7]=-0.035165
xl[8]=0.533333	yl[8]=-0.031707
xl[9]=0.600000	yl[9]=-0.027856
xl[10]=0.666667	yl[10]=-0.023754
xl[11]=0.733333	yl[11]=-0.019492
xl[12]=0.800000	yl[12]=-0.015120
xl[13]=0.866667	yl[13]=-0.010647
xl[14]=0.933333	yl[14]=-0.006047
xl[15]=1.000000	yl[15]=-0.001260

The above mentioned coordinates are used for drawing airfoil profiles. Here as an example NACA-4415 airfoil is considered. In the same manner coordinates for other four digit airfoils can be obtained from above source code and using the obtained coordinates desired profiles can be designed.

5. Resultant airfoil profile

Various design softwares can be used for design purposes like autocad,solidworks etc [5]-[9]. In the present case solidworks software is considered.A comparison between the designed airfoil profile and the airfoil profile obtained from available online sources is also shown in figure.
NACA-4415 airfoil is choosed for drawing and showing airfoil profile.

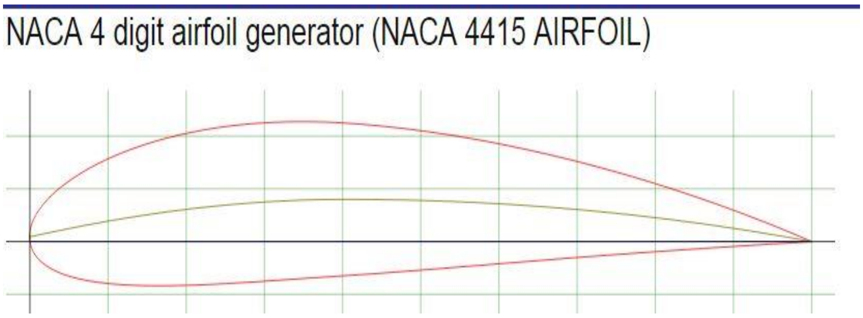


Fig 5.1.Airfoil Profile (NACA 4415) obtained from available online sources

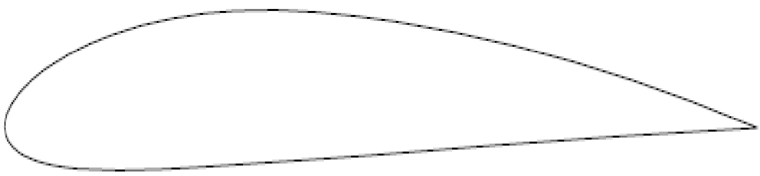


Fig 5.2.Airfoil Profile (NACA 4415) obtained from solid works using coordinates shown in the previous tables of NACA-4415.

From the above two airfoil profile it can be observed that same airfoil profile which is available in online sources can be obtained using solid works.

6. Conclusions

The detail procedure of designing the airfoil profile for NACA four digit airfoil series has been shown. Final airfoil profile can be obtained after completing some stages. Firstly ,collecting necessary equations. Secondly, write down the necessary source code for generating coordinates using any of the available programming languages. Finally, using the obtained coordinates final airfoil profile can be drawn with the help of available drawing softwares. This obtained profile now can be used for constructing the practical models.

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