Analysis of an NACA 4311 Airfoil for Flying Bike

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Abstract

The development of the wing has been always such that it should be able to produce the maximum lift due to the high pressure on the bottom surface and low pressure on the top surface of an airfoil. And these concepts clears that the flow of air/velocity of air will be low on the lower surface and higher on the upper surface of an airfoil. So, due to these differences in pressures and velocity the aerial can produce lift. Here to let fly the Bike in the air the Flat bottomed Airfoil has been chosen and usually the flat bottomed airfoil is called as the Clark Y and this has the feature as Maximum thickness of 11.7

Index terms— NACA 4311 airfoil, flat bottomed airfoil, javafoil, clark Y.

1 I. Introduction

he wing considered is the flat Bottom (NACA 4311) which is a Clark Y type usually called just because it comes under the Flat bottomed surface airfoil and has the features of maximum thickness (t/c): 11.63% @ 30.81%and maximum camber of 3.54% @ 34.52% (when plotted for 81 points) And as in order to provide the maximum lift with minimum drag we will analyze the various kinds of airfoil using the airfoil analysis software called JAVAFOIL. And the main purpose of JAVAFOIL is to determine the lift, drag and the moment characteristics of airfoils. For this reason it uses a potential flow analysis module which is based on the higher order panel method (linear varying vorticity distribution), Since the drag force is referred as the energy loss property, so to minimize it, we will choose various airfoils to compare the best one. So, with the help of JAVAFOIL we will look over the various properties and characteristics of an airfoil. a) Reason for the choosing of Clark Y type Airfoil is as follows: i. Characteristics of Clark Y:

? Clark Y has a flat bottomed profile of an airfoil and is usually safe for gliding with lower pitch in the air. ? Polikarpov R-5 ? Spirit of St. Louis ? Stinson Reliant ? Waco UPF-7 ? Currie Wot Clark YH ? Hawker Hurricane ? Ilyushin Il-2 ? Mikoyan-Gurevich MiG-3 ? Miles Magister ? Nanchang CJ-6 ? Polikarpov I-153 ? Stolp SA-900 V- Star ? Yakovlev Yak-18T

Here with the help of an Airfoil tool generator we can construct any profile of required data and can be experimented for results. Therefore, to analyze the airfoil for its characteristics and performance, a JAVAFOIL has been used which is an Aerodynamic software Source:

(http://www.airfoiltools.com/airfoil/naca4digit) for the illustration of various aerodynamic properties.

2 c) Geometry

This is the first step in JAVAFOIL to obtain the required shape of an airfoil by giving the details of airfoil or by giving the coordinates and the airfoil will be developed selecting the create airfoil option. ? Here from the above (figure **??**) we see that a graph is plotted for the airfoil and the upper surface is having the coordinates in negative mostly just because airfoil is experiencing a negative pressure and the lower surface is having a positive coordinates mostly just because it is experiencing a positive pressure which is responsible for the lift of an airfoil. Therefore, the analysis on the velocity provides the information about the behavior of the airfoil which varies with the angle of attack. Hence from the above figure of Velocity distributions we can see that how it has behaved along the length of an airfoil for different angles, Also we can see the coefficient of lift (?? ??)and Coefficient of drag (?? ??) along with the pitching moment (?? ??), coefficient of pressure (?? ??) and Mach number (?? ????).

So, here we get the velocity distribution over airfoil (NACA 4311) for 10? of angle of attack in 10 steps which is shown by the ten upper line and ten lower line indicated on the right hand side top corner of the figure 5.While the (0-0) is the velocity distribution on the surface, where we can see that the velocity distribution is low at the stagnation point as it had dropped downwards due to the high pressure and again the velocity is much high in the upper surface than lower surface and it has again dropped down in the trailing edge without overlapping of upper and lower velocity distribution profile and also it suggest that it is a laminar flow since no overlapping of profile is noticed. And the coefficient of lift (?? ??) and drag (?? ??), pitching moment(?? ??), and critical coefficient of pressure (?? ??) are increasing for every 10? angle of attack. Rather the Mach number(?? ????) is decreasing for every 10? angle of attack.

While, M 0.25 (Nm) is the pitching moment at 25% chord point. Therefore from the figure **??**, we can see the pressure coefficient in a thin red lines for ten different angle of contact. And the Critical mach number for 0? is 0.702 and for the 10? the mach number 0.400. Hence the mach no is less than 0.8 so it concludes that the flight is subsonic. While the pressure are low in the upper surface of airfoil and high on the lower surface which creates the lift.

3 f) Mach Number

Mach number (M or Ma) is the ratio of speed of an object moving through a fluid and the local speed of sound.

Where, v is the velocity of the source relative to the medium and v sound is the speed of sound in the medium. ? One can compare the velocity distribution for any angle of attack without and with ground in effect.

4 h) Flowfeild

Here in (Figure 9) the flow can be seen around the airfoil considering the angle of attack as 10? and with the boundary layer around an airfoil, it also incudes the friction to show the boundary layer to result the exact behaviour of an airfoil as in practical. Where the rectangular grid is showing the local velocity points. And these calculation uses the vorticity distribution on the surface and neglects friction which leads to no seperation flow or a wake behind the airfoil. And the streamlines are calculated from the software with the help of Runge Kutta method and Streamlines around the submersed airfoil can be seen through the blue continuity lines, while the black tuffs are the black discontinued dashes. 11 : (The velocity ratio is zero at the Red location for which the v/V is given as 0.0 at the stagnation point) j) Pressure Distribution It has been determined that as air flows along the surface of a wing at different angles of attack there are regions along the surface where the pressure is negative, or less than atmospheric, and regions where the pressure is positive, or greater than atmospheric. This negative pressure on the upper surface creates a relatively larger force on the wing than is caused by the positive pressure resulting from the air striking the lower wing surface. While the pressure distribution is described in terms of Pressure coefficient and from the figure we can see the positive pressure and negative pressure along the length of an airfoil. Because the velocity of the flow over the top of the airfoil is greater than the free-stream velocity, the pressure over the top is negative.

Therefore here (from figure 13), we have the centre of pressure at the yellow point/region and we can read the pressure as Coefficient of pressure as (-2.0), similarly we can read the positive pressure which is responsible for the lift of an airfoil as Cp= 1.0 indicated in blue color while the negative pressure can be read which is around the upper surface of an airfoil.

5 k) Boundary Layer

The boundary layer analysis describes the behaviour of an airfoil around it with the flow of air. The boundary layer module works best in the Reynolds number regime between 500'000 and 20'000'000. During the way towards the trailing edge, the method checks, whether transition from laminar to turbulent or separation occurs. The graph above shows the effect of lift over drag coefficient. Starting with infinite aspect ratio (aspect ratio = 0 on the Options card). It can be clearly seen, that for five Reynolds number (Re) the lift is increasing for larger value of (Re). As the lift will be maximum if the flow of air around the airfoil will be maximum.

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7 l) Polars for Constant Wing Loading

The lift coefficient of any body depends on the speed because the wing loading is usually fixed during flight -flying at low lift coefficients results in high speeds (and high Reynolds numbers) and vice versa. Therefore the operating points during flight would slice through a set of polars having constant Reynolds numbers. It is possible to create polars more closely related to the conditions during flight. This would require adjusting the wind speed to each lift coefficient, which is cumbersome and expensive in a wind tunnel, but feasible in a numerical tool like J AVAFOIL. And here we use the Aircraft card to calculate polars for a given wing loading.

8 n) Option

The aspect ratio is used for an approximate correction of the results on the Polar and Aircraft cards for a finite wing.

9 Global

10 Conclusion

From the analysis program in Java Foil for an NACA 4311 it is observed that on the final loading of both front and rear wings, the result is positive and there is no drop in coefficient of lift for angle of attack considered (?=10?) with the consideration of ground effect with a air density of 1.2210 kg/m? and kinematic viscosity (??) of which results for the unbounded flow for the swipe angle of 0.0 because the wing considered is uniform in cross section (rectangular) behaving under speed of sound (a=340.29 m/s) as it result the mach number.

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1


Figure 1: Fig 1 :?

2


Figure 2: Fig 2 :

3

Airfoil	Operational	No roughness	Roughness	Difference	TSR	$C_{P,\max}$	Wind speed m/s
Operational	200 kW				4.4	0.32	8.5
NACA 4312		235 kW	210 kW	15 %, 5 %	4.4, 4.4	0.36, 0.34	8.5, 8.5
CLARK Y		218 kW	202 kW	9 %, 1 %	4.4, 4.4	0.33, 0.33	8.5, 8.5

Figure 3: Fig 3 :

5

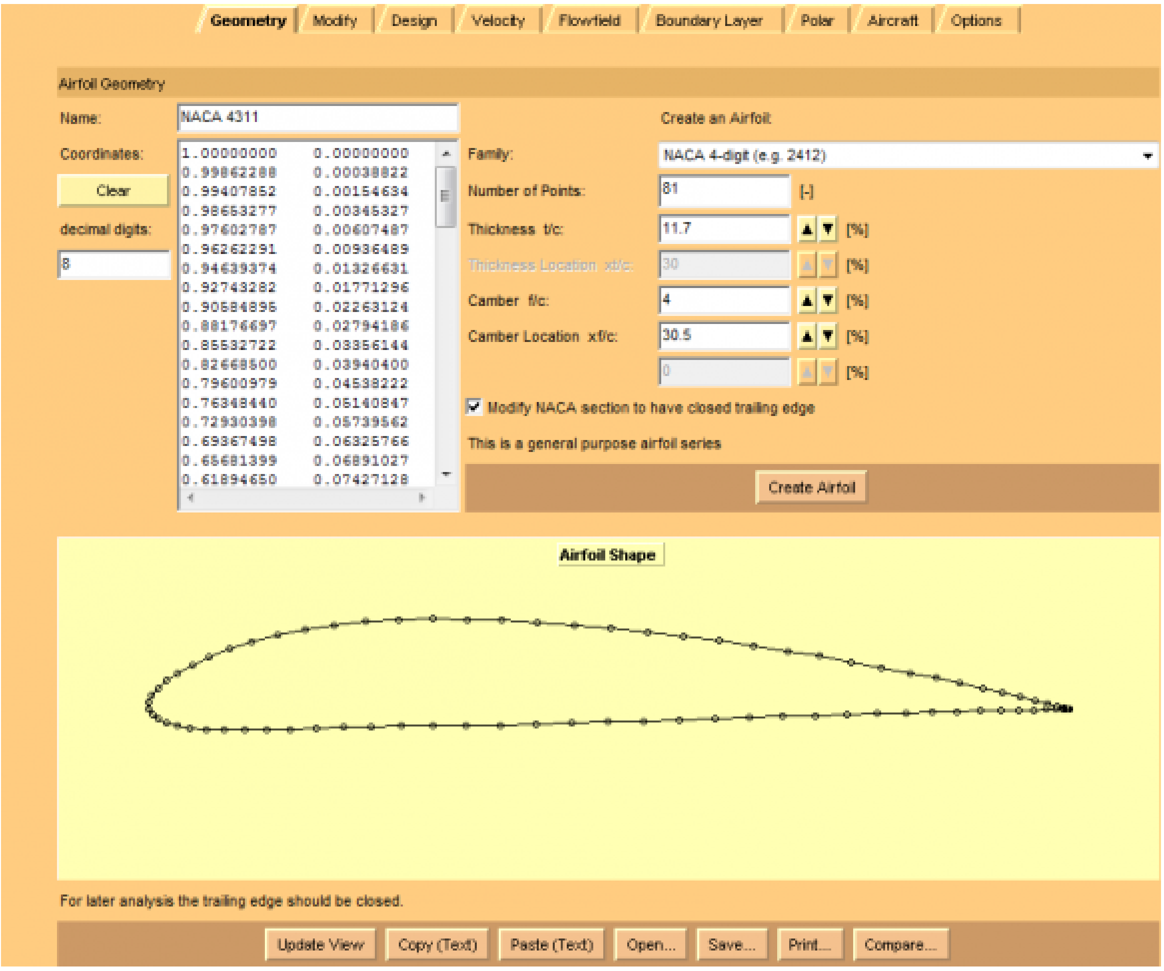

Figure 4: Fig. 5 :

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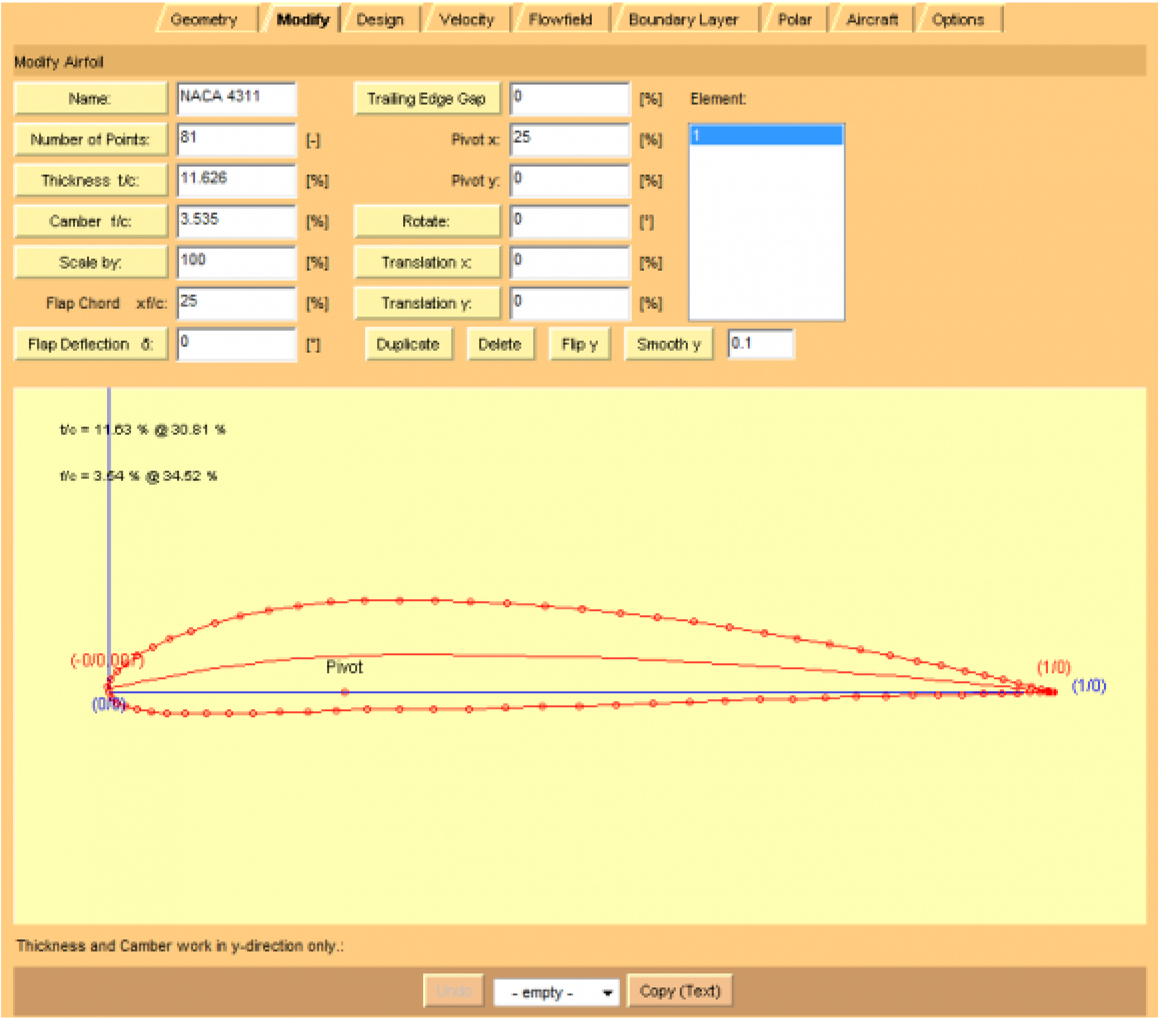


Figure 5: Fig 6 :

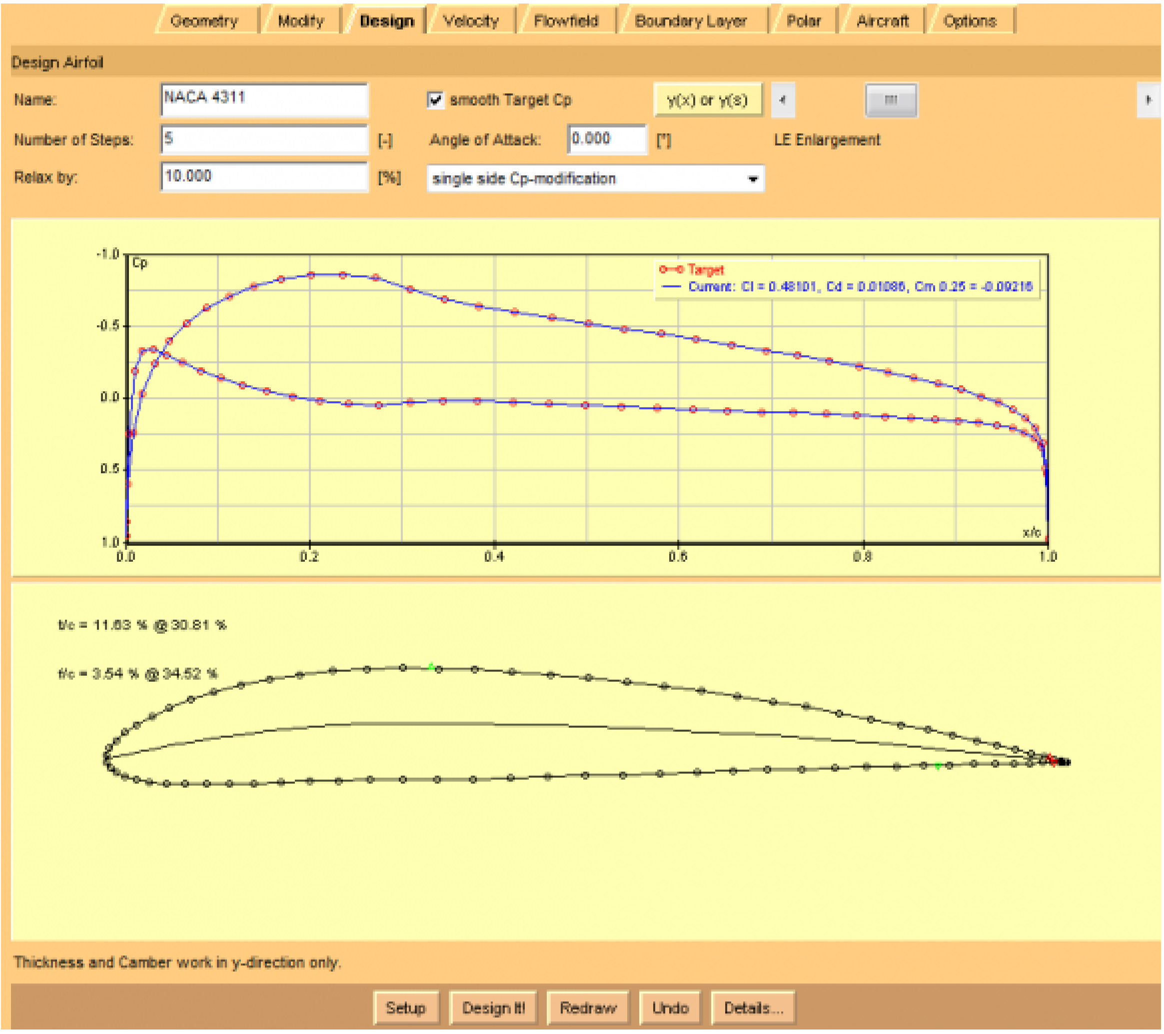
8 

Figure 6: Fig. 8 :

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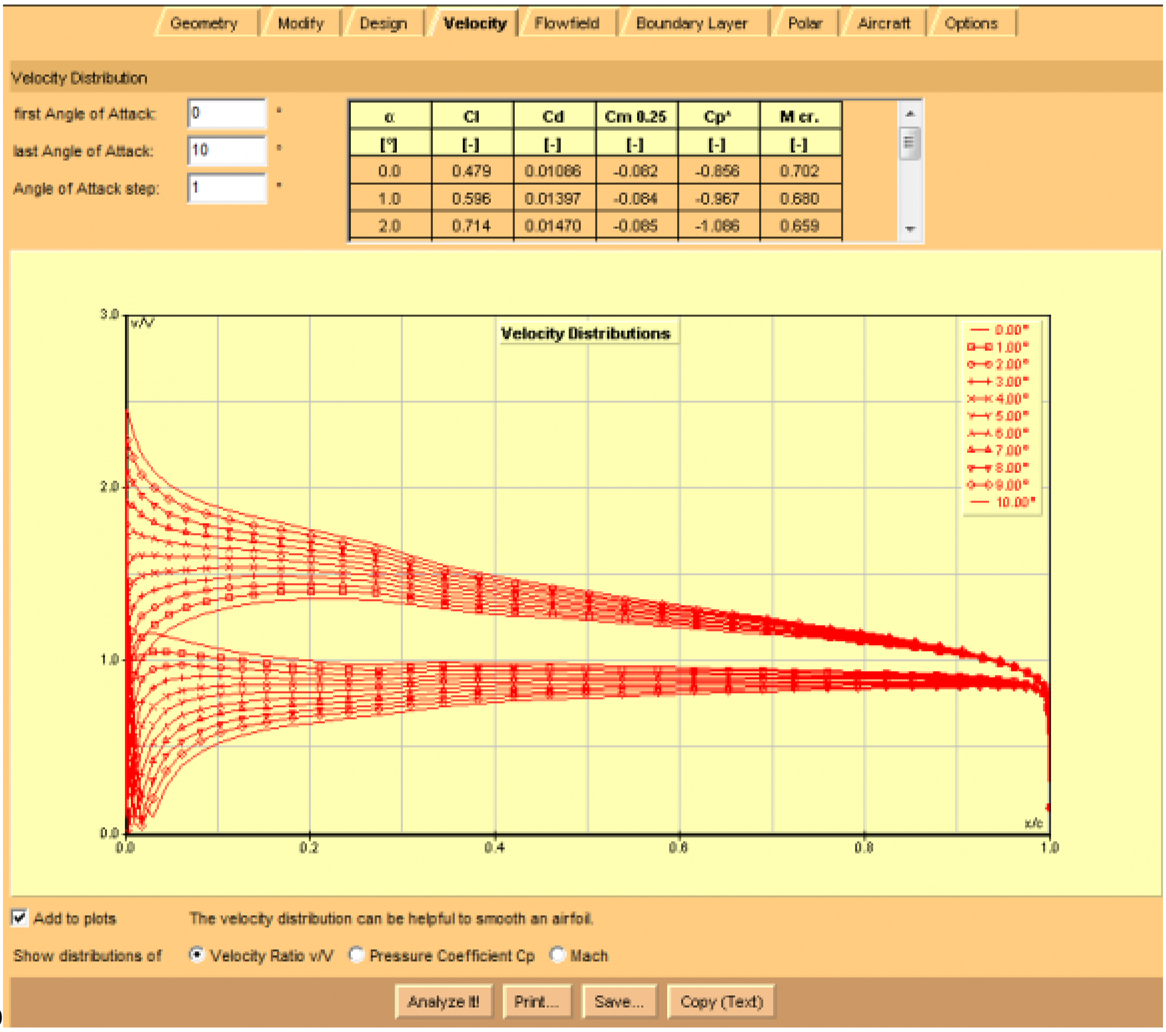
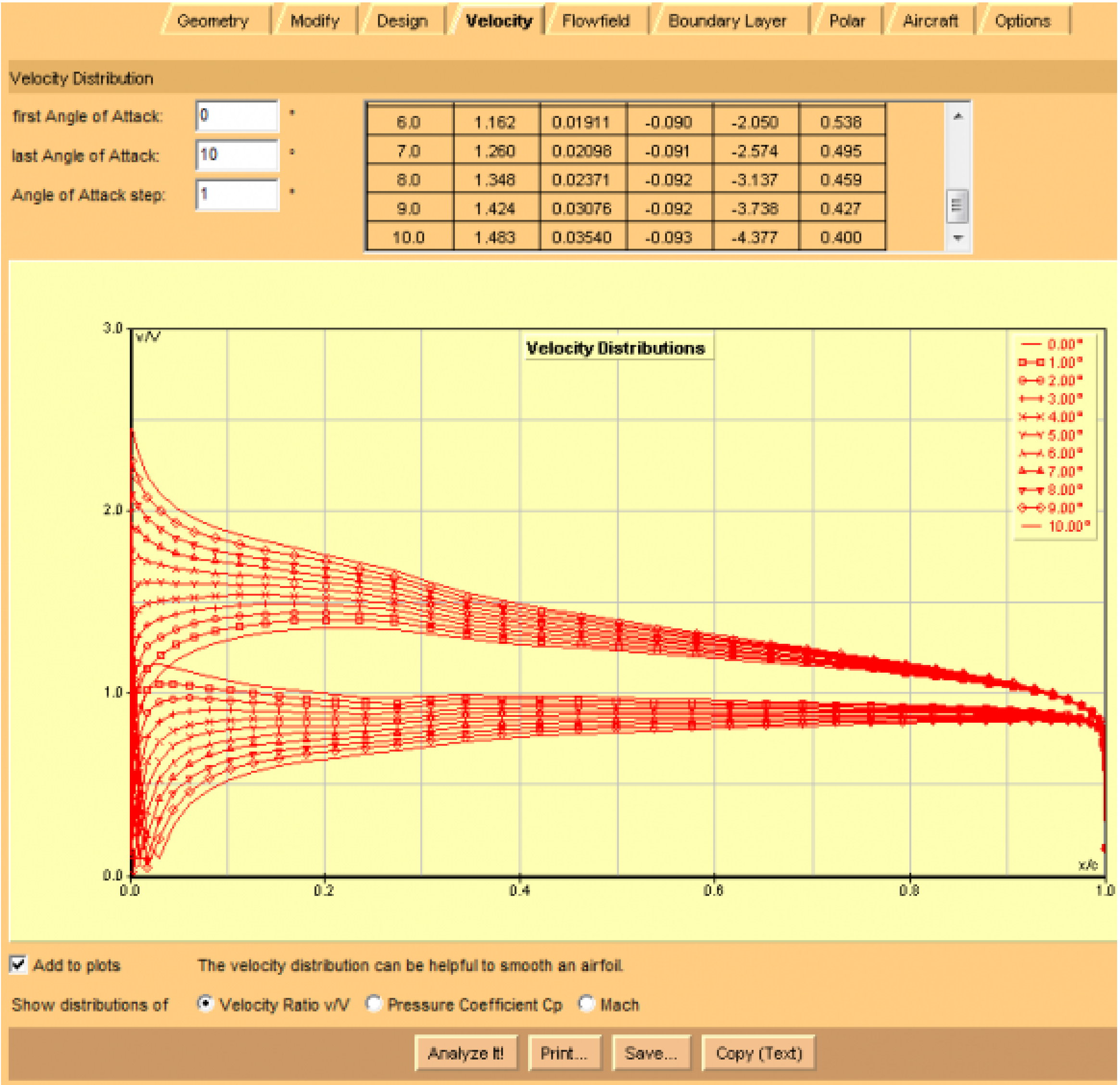


Figure 7: Fig. 9 :



12

Figure 8: Figure 12 :

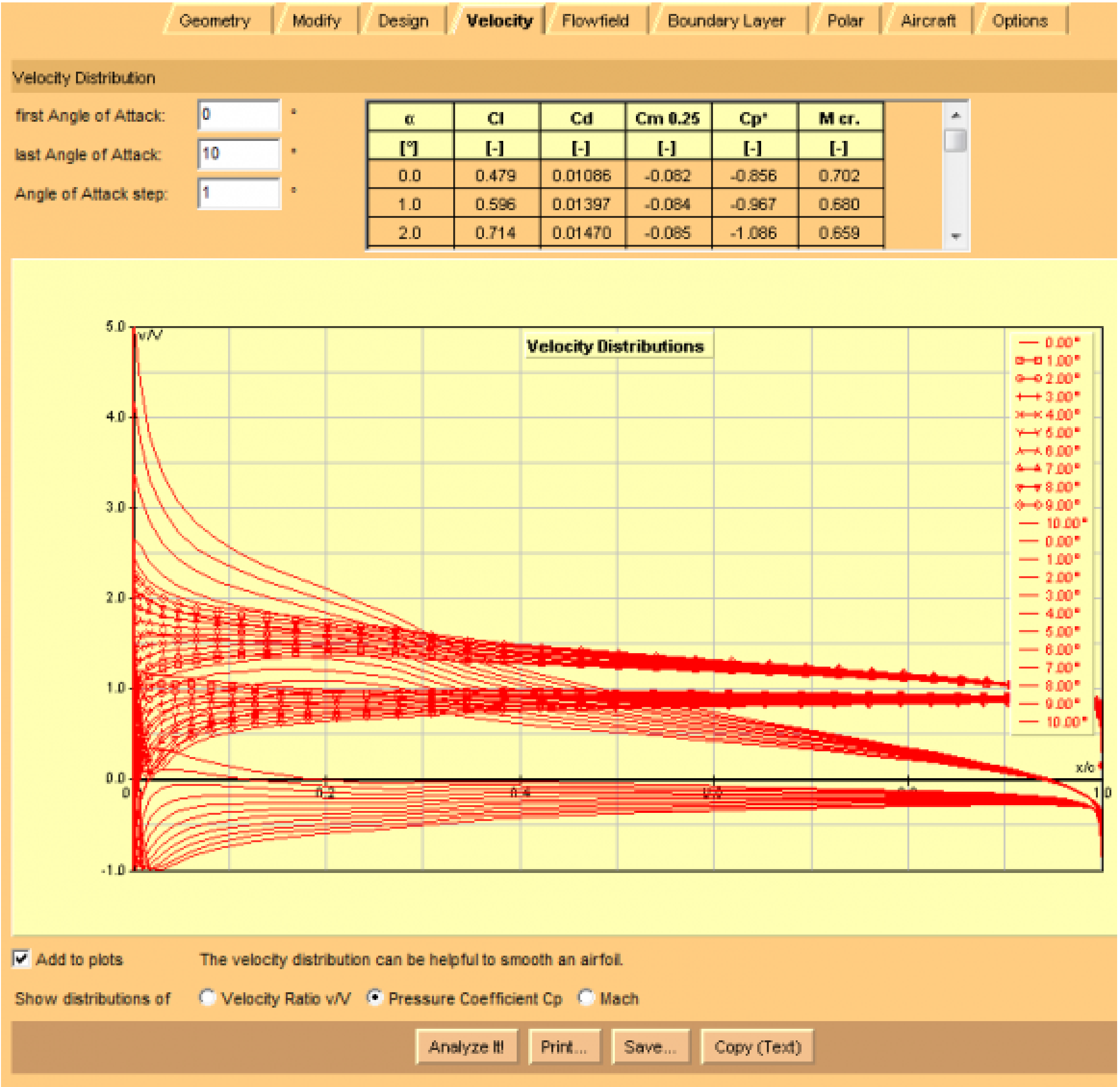


Figure 9:

14151617 $$M = \frac{v_{\text{object}}}{v_{\text{sound}}}$$

Figure 10: Fig. 14 :Fig. 15 :Fig. 16 :Fig. 17 :

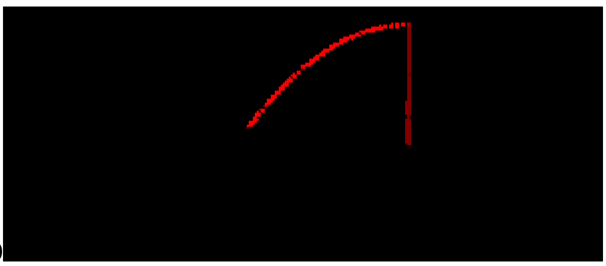
19 

Figure 11: Fig. 19 :

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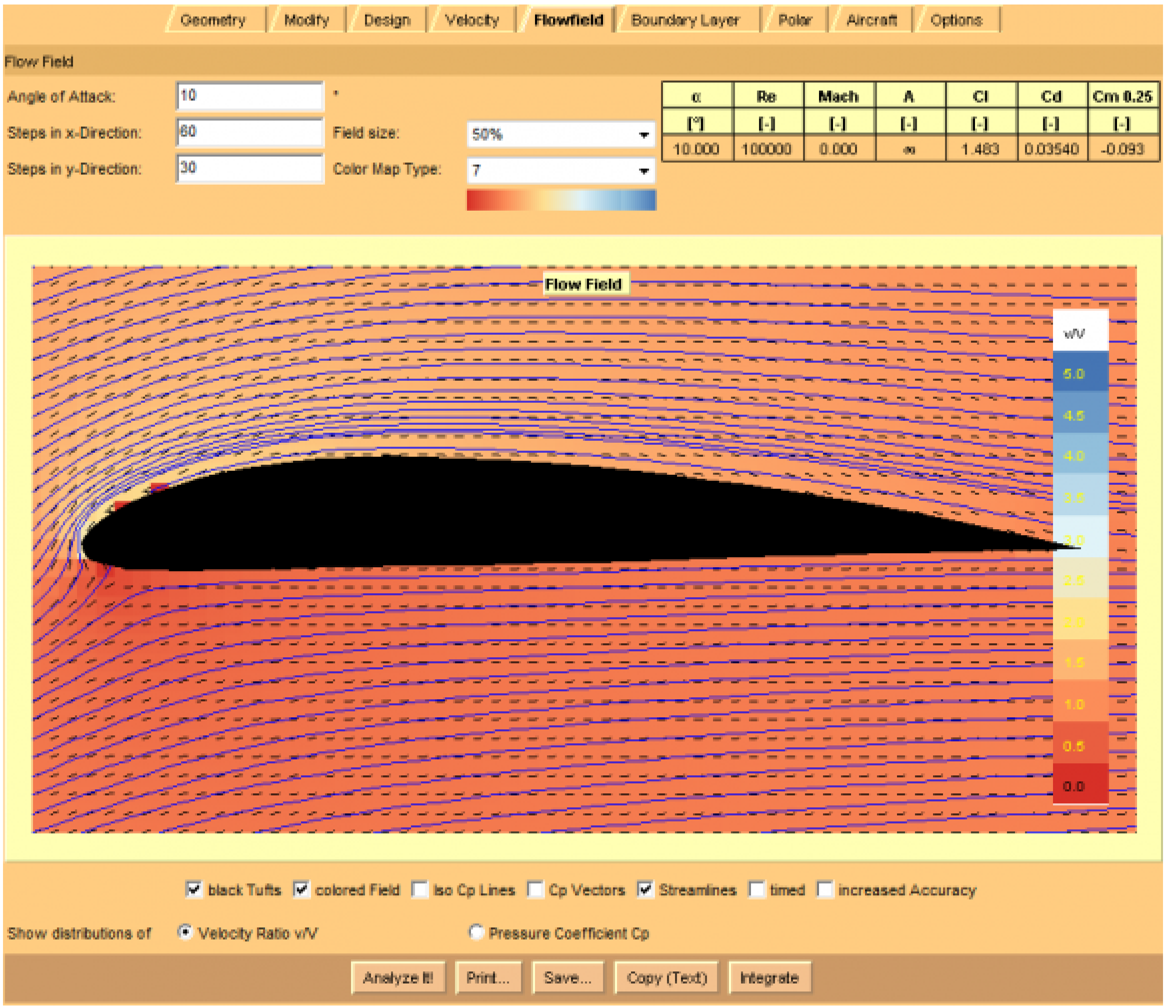
Figure 12: Fig. 20 :



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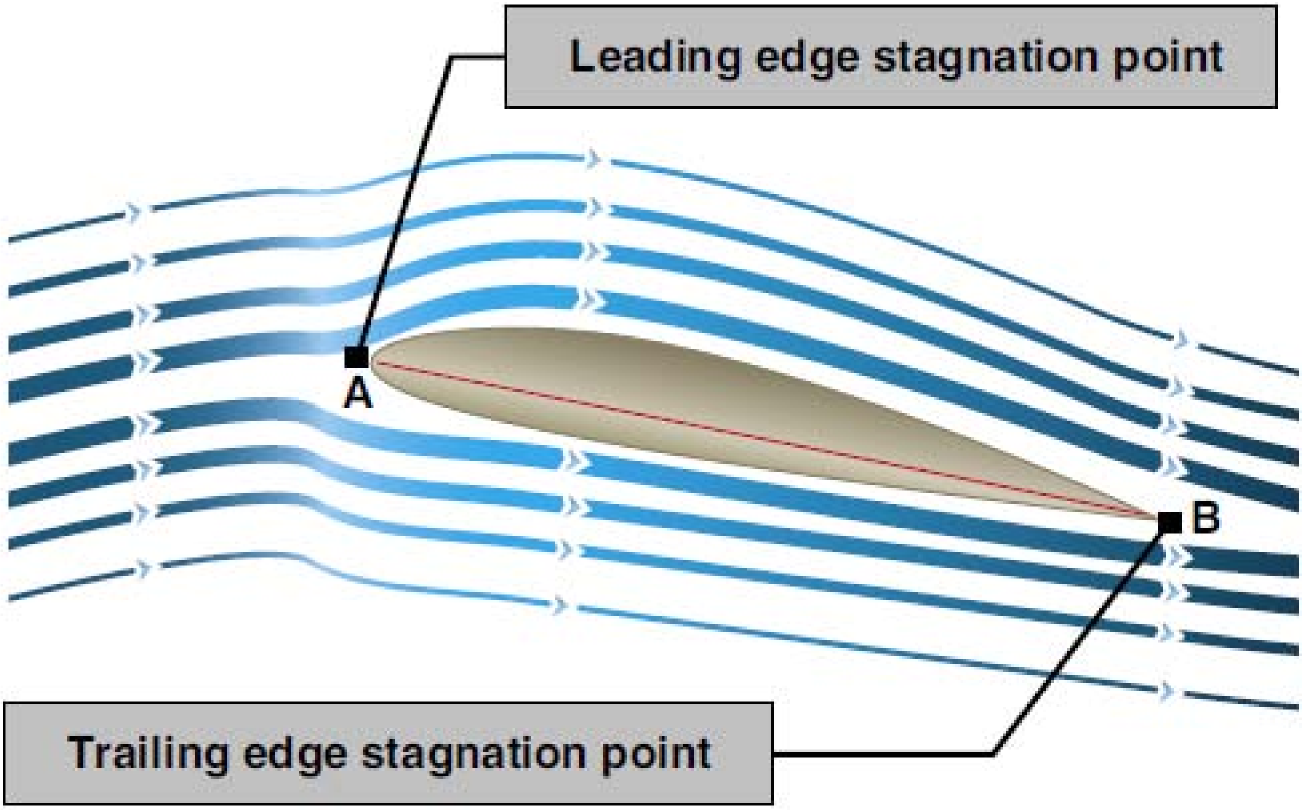

Figure 14:

110 [1]

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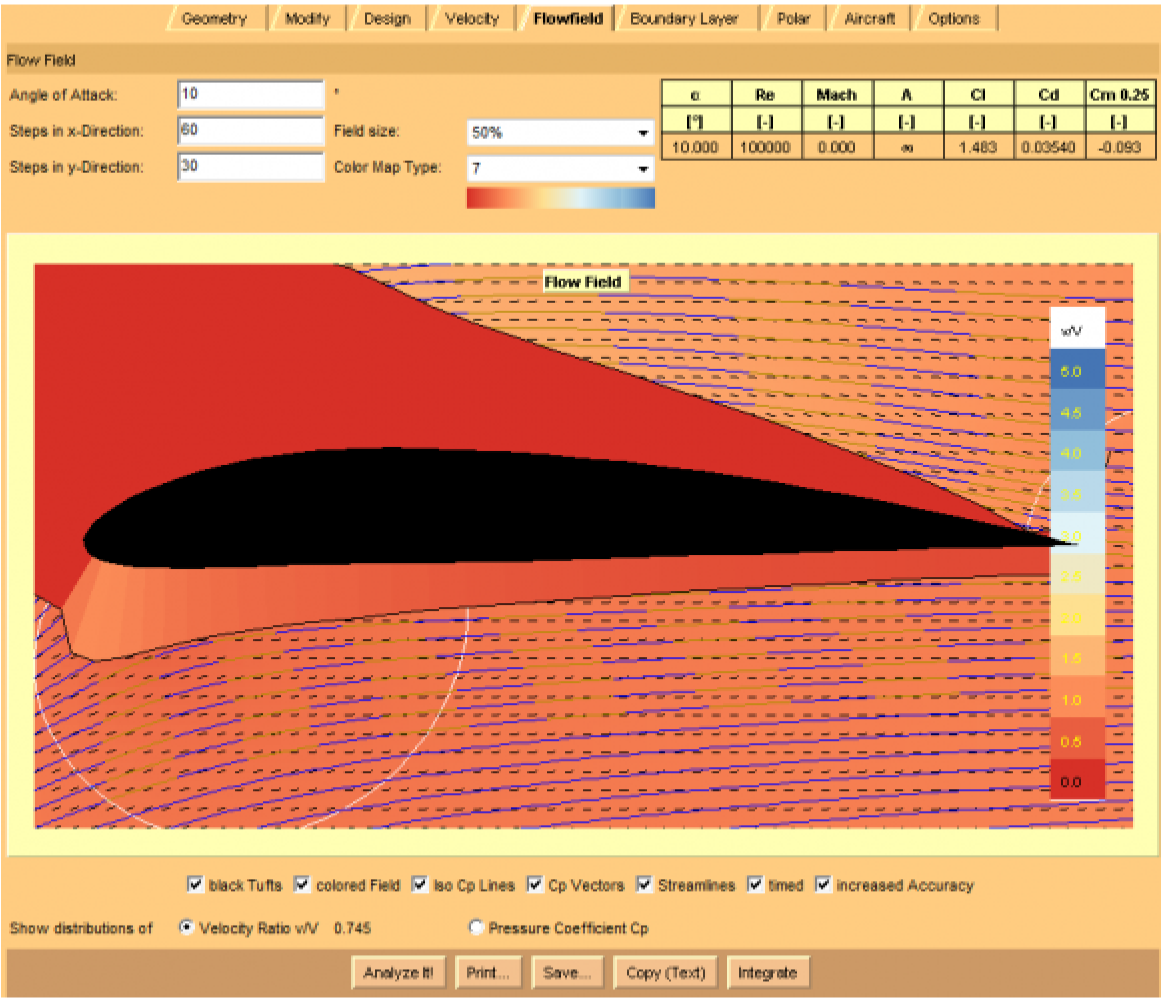

Geometry
Modify
Design
Velocity
Flowfield
Boundary Layer
Polar
Aircraft
Options
Flow Field
Angle of Attack: 10 °
Steps in x-Direction: 60
Steps in y-Direction: 30
Field size: 50%
Color Map Type: 7
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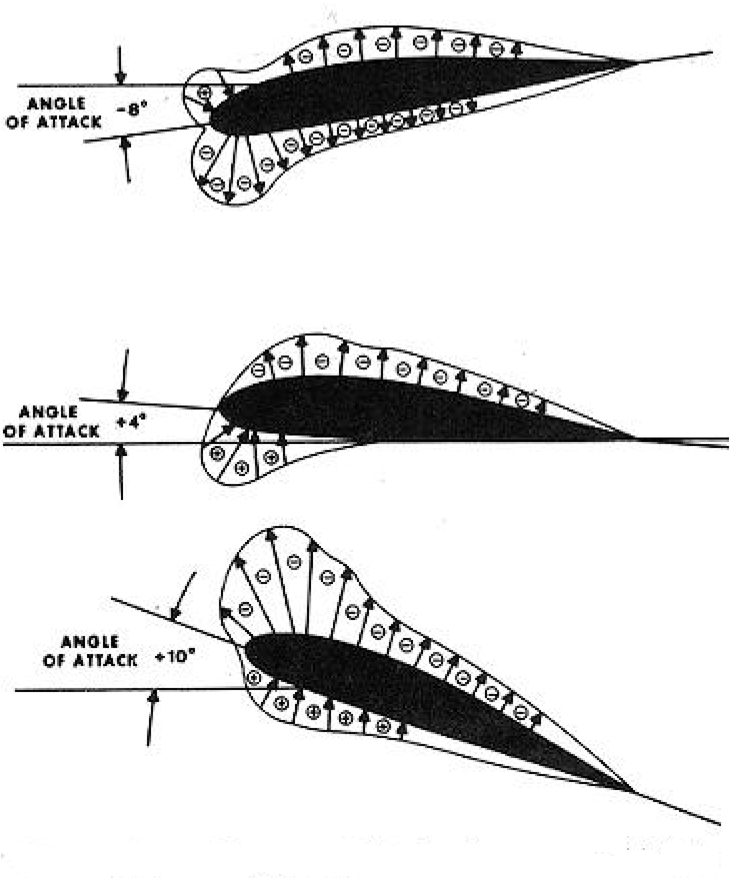
Re [-] 100000
Mach [-] 0.000
A [-] ∞
Cl [-] 1.483
Cd [-] 0.03540
Cm 0.25 [-] -0.093
Flow Field
v/V
5.0
4.5
4.0
1.0
0.5
0.0
black Tufts
colored Field
Iso Cp Lines
Cp Vectors
Streamlines
timed
increased Accuracy
Show distributions of
Velocity Ratio v/V 0.745
Pressure Coefficient Cp
Analyze It!
Print...
Save...
Copy (Text)
Integrate

Figure 15:



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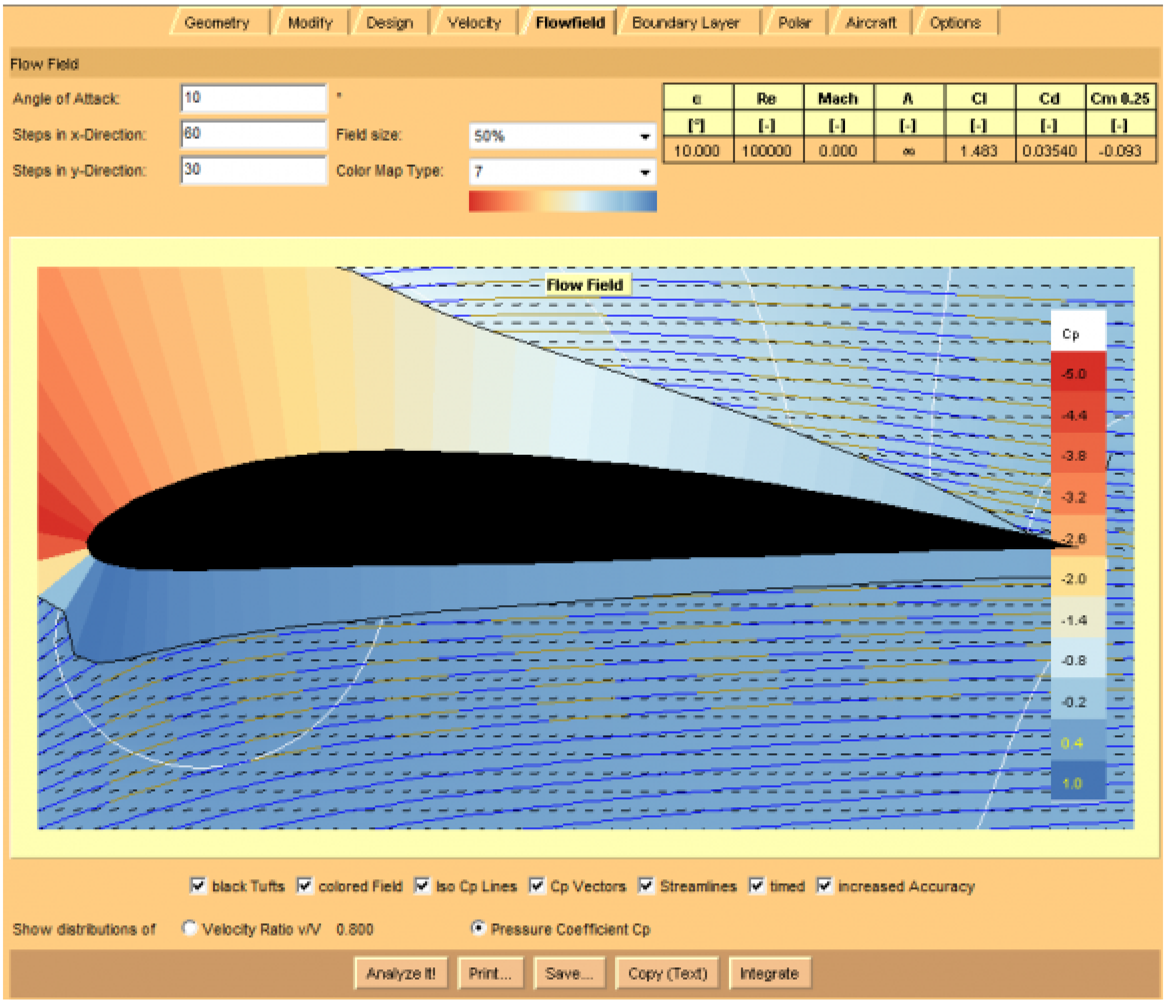
Geometry
Modify
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Flowfield
Boundary Layer
Polar
Aircraft
Options
Flow Field
Angle of Attack: 10 °
Steps in x-Direction: 60
Steps in y-Direction: 30
Field size: 50%
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Mach [-] 0.000
Λ [-] ∞
Cl [-] 1.483
Cd [-] 0.03540
Cm 0.25 [-] -0.093
Flow Field
Cp
-5.0
-4.4
-3.8
-3.2
-2.6
-2.0
-1.4
-0.8
-0.2
0.4
1.0
black Tufts
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Figure 17:

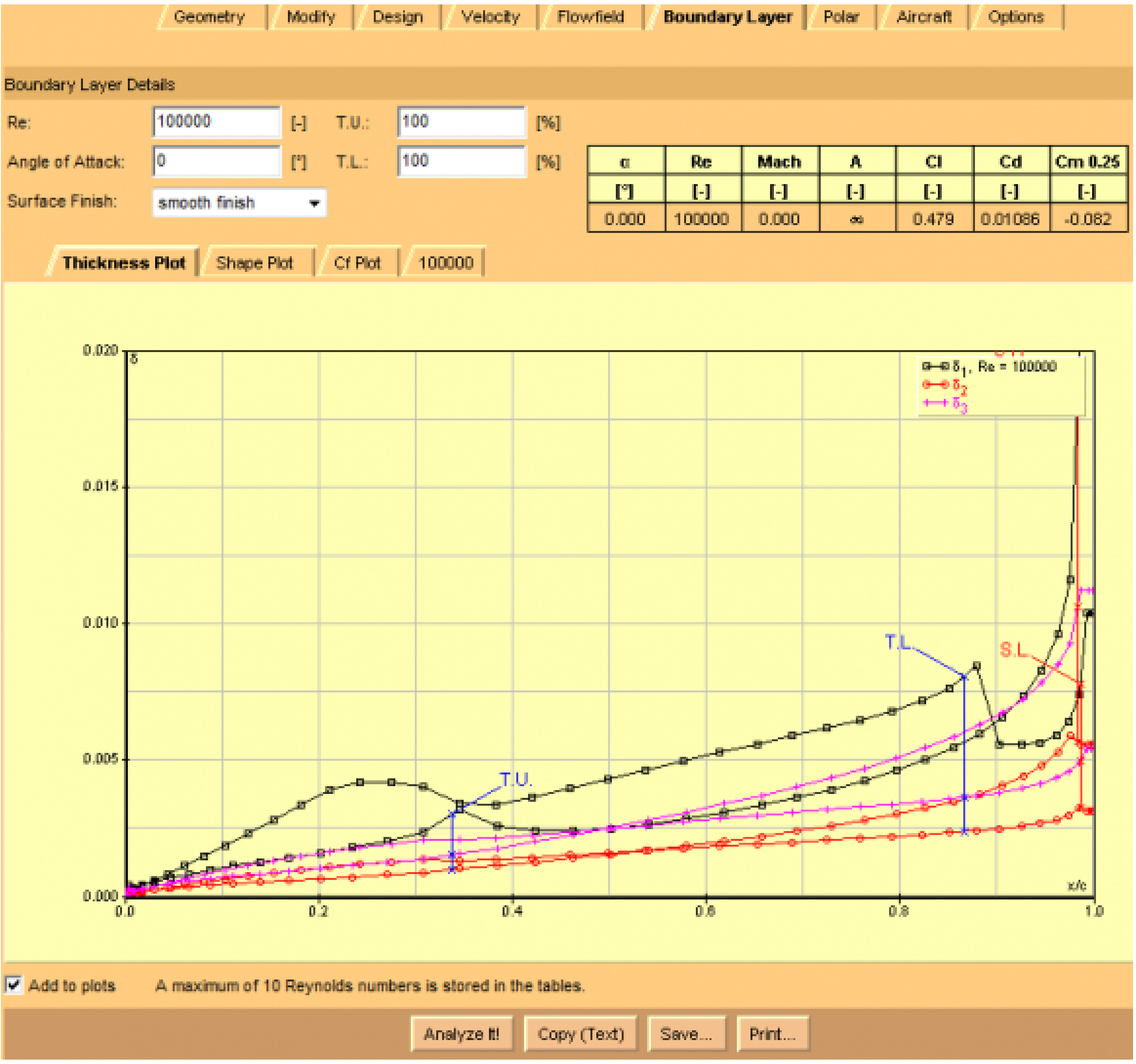

Figure 18:

δ_1

Figure 19:

u_0

Figure 20:

δ_2

Figure 21:

3

Thus, on comparing the above table 1, 2 and 3,
we have the best result from NACA 4311 due to the
modification of Clark Y type airfoil for maximum lift and
minimum drag.
b) Analysis of NACA 4311

Figure 22: Table 3 :

4

Regime: Mach Mph km/h m/s	General plane characteristics
Subsonic <0.8 <610 <980 <270	

[Note: Most often propeller-driven and commercial turbofan aircraft with high aspect-ratio (slender) wings, and rounded features like the nose and leading edges.]

Figure 23: Table 4 :

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