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% at 28% chord and maximum camber of 3.4% at 42% chord.

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Strictly as per the compliance and regulations of:
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I. Introduction

The wing considered is the flat bottomed (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.

II. Methodology

a) Considering the type of airfoil for analysis on

- NACA 4311 (Flat Bottomed Airfoil)
- NACA 3310 with thickness: 38.6%, (Flat Bottomed Airfoil)
- NACA 3310 with thickness: 31.8%, (Flat Bottomed Airfoil)
- NACA 2306, (Flat Bottomed Airfoil)
- NACA 2206, (Flat Bottomed Airfoil)
- NACA 2412, (symmetrical Airfoil)

Here with the help of an Airfoil tool generator we can construct any profile of required data and can be experimented for results. The five Flat bottom airfoil (NACA-4311, 3310, 3310 with P= 38.6%), 2306, 2206 and Symmetrical airfoil NACA 2412 are generated through this software (Airfoil tool generator) Source: http://www.airfoiltools.com/airfoil/naca4digit
On analyzing the above airfoil (a-f) in JAVAFOIL, we have the result as:

**Table 1**

<table>
<thead>
<tr>
<th>Sl. no</th>
<th>Airfoil</th>
<th>Coefficient Of Lift</th>
<th>Coefficient Of drag</th>
<th>Coefficient Of moment</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.</td>
<td>NACA 4311</td>
<td>0.46101</td>
<td>0.01089</td>
<td>-0.09216</td>
</tr>
<tr>
<td>2.</td>
<td>NACA 3310 (p=38.6%)</td>
<td>0.41505</td>
<td>0.00978</td>
<td>-0.08836</td>
</tr>
<tr>
<td>3.</td>
<td>NACA 3310 (p=31.8%)</td>
<td>0.39486</td>
<td>0.01063</td>
<td>-0.07784</td>
</tr>
<tr>
<td>4.</td>
<td>NACA 2306</td>
<td>0.22477</td>
<td>0.00958</td>
<td>-0.04518</td>
</tr>
<tr>
<td>5.</td>
<td>NACA 2206</td>
<td>0.21175</td>
<td>0.00955</td>
<td>-0.03669</td>
</tr>
<tr>
<td>6.</td>
<td>NACA 2412</td>
<td>0.25889</td>
<td>0.01032</td>
<td>-0.05525</td>
</tr>
</tbody>
</table>

WHILE FOR CLARK Y (from JAVAFOIL) we have the result as:

**Table 2**: (Javafoil analysis)

<table>
<thead>
<tr>
<th>Sl.no</th>
<th>Airfoil</th>
<th>Coefficient Of Lift</th>
<th>Coefficient of drag</th>
<th>Coeff Of moment</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.</td>
<td>Clark Y</td>
<td>0.44560</td>
<td>0.01231</td>
<td>-0.09714</td>
</tr>
</tbody>
</table>

**Table 3**: (Result from Gedser Simulation), A textbook on the thesis in Aeronautical Engineering

<table>
<thead>
<tr>
<th>Airfoil</th>
<th>Operational</th>
<th>No roughness</th>
<th>Roughness</th>
<th>Difference</th>
<th>TSR</th>
<th>Cpl/blur</th>
<th>Wind speed m/s</th>
</tr>
</thead>
<tbody>
<tr>
<td>NACA 4312</td>
<td>235 kW</td>
<td>210 kW</td>
<td>15, 5%</td>
<td>4, 4</td>
<td>4.4</td>
<td>0.32</td>
<td>8.5</td>
</tr>
<tr>
<td>CLARK Y</td>
<td>218 kW</td>
<td>202 kW</td>
<td>9, 1%</td>
<td>4, 4</td>
<td>4.4</td>
<td>0.33</td>
<td>8.5, 8.5</td>
</tr>
</tbody>
</table>

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**

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

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.
Fig 2: Geometry card: (here we observe the required airfoil in 2d view in a scale of 1/1)
So, after modification we get properties of aifoil on modified screen are

- Smooth Y = 0.1, which describes that the airfoil has a smooth spline curve.
- (Pivot x=25%) horizontally at red point describes that the angle of attack of the airfoil is always change by rotating the section around the pivot point specified on the Modify card.
d) **Design**

![Design Airfoil](image)

**Fig 4:** (Here we can see the 2D Dimensional design of the NACA 4311 Airfoil, and it is delivering a lift of (Coefficient of lift) = 0.48101 and Coefficient of drag as 0.01086 at an angle of attack = 0°, while the graph shows the coefficient of pressure along the length of the chord(c))

- Here from the above (figure 4) 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.

While, $\frac{L}{D} = \frac{C_l}{C_d}$ ratio gives Glide Ratio of the flight
e) Velocity

After design first it will calculate the distribution of the velocity on the surface of airfoil which can be integrated to get the lift and the moment coefficient. Number for different angle of attack.

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 \( C_l \) and Coefficient of drag \( C_d \) along with the pitching moment \( C_m \), coefficient of pressure \( C_p \) and Mach number \( M_c \).

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 \( C_l \) and drag \( C_d \), pitching moment \( C_m \), and critical coefficient of pressure \( C_p \) are increasing for every 10° angle of attack. Rather the Mach number \( M_c \) is decreasing for every 10° angle of attack.

While, \( M_{0.25} \) (Nm) is the pitching moment at 25% chord point.
Fig 6: (Velocity distribution for $10^\circ$ angle of attack with different characteristics of ($C_l$, $C_d$, $C_m$) and Mach number)
Therefore from the figure 7, we can see the pressure coefficient in a thin red lines for ten different angles of attack. And the Critical mach number for $0^\circ$ is 0.702 and for the $10^\circ$ 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.

**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.

\[
M = \frac{v_{object}}{v_{sound}}
\]

Where, $v$ is the velocity of the source relative to the medium and $v_{sound}$ is the speed of sound in the medium.

**Table 4:** (General Plane Characteristic)

<table>
<thead>
<tr>
<th>Regime</th>
<th>Mach</th>
<th>Mph</th>
<th>km/h</th>
<th>m/s</th>
<th>General plane characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subsonic</td>
<td>&lt;0.8</td>
<td>&lt;610</td>
<td>&lt;980</td>
<td>&lt;270</td>
<td>Most often propeller-driven and commercial turbofan aircraft with high aspect-ratio (slender) wings, and rounded features like the nose and leading edges.</td>
</tr>
</tbody>
</table>
Fig. 8: Mach number in transonic airflow around an airfoil; M < 1 (a) and M > 1 (b)

(g) Thus from Figure 5, 6 & 7
- One can compare the velocity distribution for any angle of attack without and with ground in effect.

(h) Flowfield
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 separation 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 submerged airfoil can be seen through the blue continuity lines, while the black tuffs are the black discontinued dashes.
Fig. 9: Streamlines around the submersed hydrofoil (note that image is clipped at y=0) but the generated surface wave are extending above this border.
Fig. 10: (stagnation points)
i) Stagnation Point

A stagnation point is a point in a flow field where the local velocity of the fluid is zero.

Fig. 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.
Figure 12: Pressure distribution on an airfoil

Figure 13: Pressure distribution around airfoil
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 $C_p = 1.0$ indicated in blue color while the negative pressure can be read which is around the upper surface of an airfoil.

**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.

![Analyzing the Boundary Layer of an Airfoil](image)

**Fig. 14:** Analyzed boundary layer of NACA 4311

Therefore (from figure 14), we see that for $\delta_1$, $\delta_2$ and $\delta_3$, the blue line is indicating transition of flow from laminar to turbulent on the upper layer of the airfoil surface (TU) and transition of flow from laminar to turbulent on the lower layer of the airfoil surface (TL) while (SL) is indicating the turbulent separation of the flow near the end of the trailing edge.

Where,

- $\delta_1 (m)$ is the displacement thickness of boundary layer is the distance by which a surface would have to be moved in the direction perpendicular to its normal vector away from the reference plane in an inviscid fluid stream of velocity $U_0$ to give the
same flow rate as occurs between the surface and the reference plane in a real fluid.

- \( \delta_2 \) (m) is momentum thickness of boundary layer
  - is the distance by which a surface would have to be moved parallel to itself towards the reference plane in an inviscid fluid stream of velocity \( u_0 \) to give the same total momentum as exists between the surface and the reference plane in a real fluid.

- \( \delta_3 \) (m) is energy thickness of boundary layer

- \( T \) is transition laminar-turbulent
- \( S \) is turbulent separation
- \( U \) is upper surface
- \( L \) is Lower surface

A shape factor is used in boundary layer flow to determine the nature of the flow.

\[
H = \frac{\delta^*}{\theta}, \text{ Note } \delta^* = \delta_1/\delta_3 \text{ and } \theta = \delta_2
\]

Where, \( H \) is the shape factor, \( \delta^* \) is the displacement thickness and \( \theta \) is the momentum thickness. The higher the value of \( H \), the stronger the adverse pressure gradient. A high adverse pressure gradient can greatly reduce the Reynolds number at which transition into turbulence may occur.

- \( H_{12} = \delta_1/\delta_2 \) is the shape factor of boundary layer and \( H_{32} = \delta_3/\delta_2 \) is the shape factor of boundary layer, \( C_f \) is the local skin friction coefficient.

**Fig. 15**: Flow state graph on airfoil NACA 4311
Pressure gradient is high (red line at point $H_{12} > 3.5$ for the Reynolds no. Here (from figure 18) it can be observed that for the maximum thickness of the airfoil, number (Re) = 100000. Also we can see the for the $H_{32}$

Where,

$H_{32} < 1.51509$, will have the laminar flow and $H_{12} < 1.46$ will have the turbulent flow, which can be observed from the figure 18, at TU, TL and SU, SL. The blue line is indicating transition of flow from laminar to turbulent on the upper layer of the airfoil surface (TU) and transition of flow from laminar to turbulent on the lower layer of the airfoil surface (TL) while (SL) and (SU) is indicating the turbulent separation of the flow near the end of the trailing edge in the lower and upper surface of NACA 4311 in the both cases of $H_{12}$ and $H_{32}$.

Table 5: Shape factor boundary layer condition

<table>
<thead>
<tr>
<th>Flow State</th>
<th>Separation assumed when</th>
</tr>
</thead>
<tbody>
<tr>
<td>Laminar</td>
<td>$H_{32} &lt; 1.51509$</td>
</tr>
<tr>
<td>Turbulent</td>
<td>$H_{12} &lt; 1.46$</td>
</tr>
</tbody>
</table>

Also shape factor displacement thickness/momentum thickness has the relation as

$$H_{12} = \frac{61}{\pi^2}$$

Fig. 16: (local skin friction coefficient)
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**Fig. 17**: (different value of calculated properties for the boundary layer)

**Fig. 18**: (Lift versus drag coefficient polars for a NACA 4311 airfoil and wings of different aspect ratio)
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.

I) 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.

![Graph showing lift over drag coefficient](image)

**Fig. 19:** (polar condition of flight for different Reynolds number (Re))

m) Aircraft

The Polars card analyzes the airfoil for constant Reynolds numbers. For an aircraft in flight the lift coefficient depends on the flight speed and hence on the Reynolds number.
Notes

- To check the airfoil for different angles of attack, one can analyze complete polar for different angles of attack and Reynolds numbers. The angle of attack is changed by rotating the airfoil around the point (0.25/0), which will change the height of the airfoils 25% chord point above ground somewhat.

n) Option

The aspect ratio is used for an approximate correction of the results on the Polar and Aircraft cards for a finite wing.
III. 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 (α=10°) with the consideration of ground effect with a air density of 1.2210 kg/m³ and kinematic viscosity (v) 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.

References Références Referencias

10. Fan wing manned aircraft project (http://www-fanwing.com/fanwing20manned%20aircraft%20project%202013.pdf)
25. Dr. Richard Eppler, Eppler Airfoil Design and Analysis Code, Petersburg, USA.
27. J. F. Marchman and Todd D Werme , Clark y airfoil performance at low Reynolds Number, Virginia polytechnic institute and state university, Blackburg, Virginia 1-7, Jan 9-12, 1984/reno, Nevada, Publisher: American institute of Aeronautics and Astronauts.
29. Prof. E.G. Tulapurkara, Flight dynamics-I: Chapter-3
33. Christian J. Kähler, Sven Scharnowski, Christian Cierpka, High resolution velocity profile measurements in turbulent boundary layers,16th Int
34. Lelanie Smith, An interactive boundary layer modelling methodology for Aerodynamic flows, Submitted in partial fulfilment of the degree Masters of Engineering Department of Mechanical and Aeronautical Engineering University of Pretoria November 2011, p 1-75.


43. Galal Bahgat Salem and Mohammed Khalil Ibrahim, Measurement of Pressure Distribution over a Cambered Airfoil, Aerospace Engineering Department, Aerodynamics Laboratory – Experiment #1, Cairo University, Faculty of Engineering, April 2004, Page 1-8.

44. Dr. Peyman Taheri NUMERICAL CALCULATION OF LIFT AND DRAG COEFFICIENTS FOR AN ELLIPSE AIRFOIL, ENSC 283 INTRODUCTION TO FLUID MECHANICS, ENSC 283 (Spring 2013), Simon Fraser University. p1-5.


47. NASA STI PROGRAM, NASA SP-7037 (303), April 1994.


