DESIGN OF AEROFOIL FOR A SINGLE SEATER HOME BASED AIRCRAFT

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ABSTRACT

First we start with design and development. Let's take the military as an example. Suppose the military wants to design new weapon systems that will be more accurate than systems they currently have. The first things they have to do is get a team of scientists and whatever other personnel are needed to design the system. Then, after they come up with a design they are pleased with they have to get the engineers to develop it. All these people either have to be hired or contracted. Some of the work maybe sub contracted.

Then there is the acquisition of the materials needed. Some of the materials may already be on hand while others have to be purchased. Some will be purchased domestically and others will need to be purchased from overseas. In many cases bids will be put in to various companies to see who can supply the materials at the cheapest cost.

After the materials are acquired there's the matter of storage. In many cases the materials is either of so large a quantity or so large in size that the storage facilities need to be specially built in order to adequately provide the needed space. This of course has to be figured in to the equation.

Then there's the matter of movement and distribution to may be the various military bases around the country or even overseas. Trucks or planes need to be acquired in order to distribute the materials if there aren't enough readymade transportation vehicles. Transportation costs alone can be astronomical.

KEY WORDS

Aerofoil – Home based air craft – design - 'C' – Software

1.0 INTRODUCTION

Three major types of airplane designs are

A. Conceptual design
B. Preliminary design
C. Detailed design

A. Conceptual design:

It depends on what are the major factors for designing the aircraft.

Power plant Location:

The Power plant location is either padded (or) Buried type engines are more preferred. Rear location is preferred for low drag, reduced shock & to the whole thrust.

Selection of Engine:

The engine should be selected according to the power required.

Wing selection:

The selection of wing depends upon the selection of

(1) Low wing
(2) Mid wing
(3) High wing

B. Preliminary design:

Preliminary is based on Loitering. 'U' is the mathematical method of skinning the aircraft, the aircraft look like a masked body.

Preliminary design is done with help of 'C' SOFTWARE.

C. Detailed design:

In the detailed design considers each & every rivets, bolts, paints etc. In this design the connection & allocations are made.

2.0 PROCEDURE:

The airfoil can be selected from the $C_{L_{max}}$ which we can determined from the calculation.

The $C_{L_{max}}$ for the airfoil has to be selected for choosing the specified airfoil.

Blade Element Theory for Airfoil:

In the previous section, we looked at momentum theory. This theory gave us four useful pieces of information:

(1) Induced velocity far downstream in the rotor wake, called downwash, is twice that at the rotor disk, called inflow.

(2) The ideal power coefficient C_p in hover equals $C_T^{3/2}/\sqrt{2}$

(3) The induced power is minimized for a given thrust coefficient, if the induced velocity in the far wake is uniform.

(4) The induced velocity at the rotor disk is related to the thrust coefficient in hover by

$$\lambda_i = \frac{v}{\Omega R} = \sqrt{\frac{C_T}{2}}$$

Momentum theory cannot help us analyze specific rotor blades, or distinguish between the number of blades and their other physical characteristics such as twist, taper, camber etc. In order to do these, we turn to a theory called **blade element theory**.

Blade element theory is similar to the strip theory in fixed wing aerodynamics. The blade is assumed to be made of several infinitesimal strips of width 'dr'. The lift and drag are estimated at the strip using 2-D airfoil characteristics of the airfoil at that strip, and what we know about the local flow magnitude, such as the angular velocity, climb speed V, and inflow v. The lift L , and drag D multiplied by the in-plane velocity of the rotor are integrated with respect to r, from root to tip to obtain the thrust T and the power P consumed by a single rotor blade. For multi-bladed rotors, this integrated expression is multiplied by the number of blades, b.

Note: Wayne Johnson uses the symbol 'N' for number of blades.

Consider a typical element or strip shown below. The blade "sees" an in-plane velocity U_T, that is tangential to the plane of rotation. The magnitude of U_T is, of course, Δr, where r is the radial position of the strip. This element has a pitch angle equal to Φ. That is, the angle between the plane of rotation and the line of zero lift is θ. If there were no climb velocity V, or induced inflow v, this would be the section angle of attack.

These two components of velocity V and v change the flow direction by amountsΔ, as shown in the figure above. Here,

$$\phi = \arctan\left(\frac{V+v}{\Omega r}\right)$$

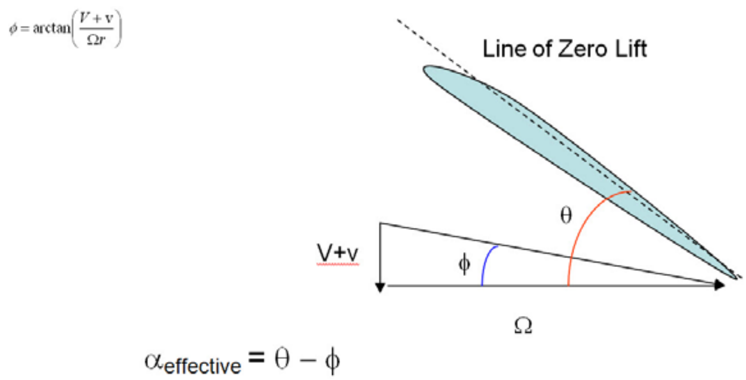
Thus, the effective angle of attack is □□ . The airfoil lift and drag coefficients C_l and C_d at this effective angle of attack may be looked up from a table of airfoil characteristics. The lift and drag forces will be perpendicular to, and along the apparent stream direction.

These forces are given by

$$L' = \frac{1}{2}\rho\left(U_T^2 + U_P^2\right)cC_l$$

$$D' = \frac{1}{2}\rho\left(U_T^2 + U_P^2\right)cC_d$$

The L' and D' have units of force per unit span. They must be rotated in directions normal to, and tangential to the rotor disk, respectively, and multiplied by the strip width dr to get the thrust and drag components, as shown below.



$$dT = \left(L'\cos(\phi) - D'\sin(\phi)\right)dr = \frac{1}{2}\rho\left(U_T^2 + U_P^2\right)c\left(C_l\cos(\phi) - C_d\sin(\phi)\right)dr$$

$$dF_x = \left(D'\cos(\phi) + L'\sin(\phi)\right)dr = \frac{1}{2}\rho\left(U_T^2 + U_P^2\right)c\left(C_d\cos(\phi) + C_l\sin(\phi)\right)dr$$

$$dP = U_T dF_x = \Omega r dF_X$$

Finally, the thrust and power T and P may be found by integrating dT and dP above from root to tip (r=0 to r=R), and multiplying the results by the total number of blades, b.

The above integration can, in general, be only numerically done since the chord c, the sectional lift and drag coefficients may vary along the span. Finally, the inflow velocity v depends on T. Thus, an iterative process will be needed to find the quantity v.

Approximate expressions for thrust and power may, however, be found if we are willing to make a number of approximations:

a) The chord c is constant,

b) The inflow velocity v and climb velocity V are small. Thus, $\theta - \phi << 1$, and $\theta - \phi << 1$. We can approximate $\cos(\theta - \phi)$ by unity, and approximate $\sin(\theta - \phi)$ by $(\theta - \phi)$.

c) The lift coefficient is a linear function of the effective angle of attack, that is, $\theta - \phi$. Thus,

$$C_l = a(\theta - \phi)$$

Where a is the lift curve slope. For low speeds, a may be set equal to 5.7 per radian.

d) C_d is small. So, $C_d \sin(\Phi)$ may be neglected.

e) The in-plane velocity U_T is much larger than the normal component U_P over must of the rotor, except near the hub.

With these assumptions, thrust T may be expressed as

$$T = \frac{1}{2}\rho cba\Omega^2 \int_{r=0}^{r=R} \left(\theta - \frac{V}{\Omega r} - \frac{v}{\Omega r}\right) r^2 dr$$

$$P = \frac{1}{2}\rho cba\Omega^3 \int_{r=0}^{r=R} \left[\left(\theta - \frac{V}{\Omega r} - \frac{v}{\Omega r}\right)\left(\frac{V}{\Omega r} + \frac{v}{\Omega r}\right) + C_d\right] r^3 dr$$

To perform the integration, we need to know how the pitch angle Φ varies with r. Many rotor blades are twisted, and it is not reasonable to assume that the pitch angle Φ is constant. Two choices are common.

Linearly Twisted Blade:

Here, we assume that the pitch angle varies as

$$\theta = E + Fr$$

where E and F are constants. Using this definition, and performing the integration (check!), we get:

$$T = \frac{b}{2}\rho\Omega^2 ac\left[\frac{1}{3}\left(E + \frac{3}{4}FR\right) - \frac{V+v}{2\Omega R}\right]R^3 = \frac{b}{2}\rho ac(\Omega R)^2 R\left[\frac{\theta_{.75}}{3} - \lambda/2\right]$$

$$C_T = \frac{abc}{2\pi R}\left[\frac{\theta_{.75}}{3} - \lambda/2\right] = \frac{a\sigma}{2}\left[\frac{\theta_{.75}}{3} - \lambda/2\right]$$

where

$$\sigma = \mathrm{solidity} = \mathrm{BladeArea}/\mathrm{DiskArea} = bc/\pi R$$

$$\lambda = \mathrm{InflowRatio} = \frac{V+v}{\Omega R}$$

Notice that the thrust coefficient is linearly proportional to the pitch angle Φ at the 75% Radius. This is why the pitch angle is usually defined at the 75% R in industry.

The expression for power may be integrated in a similar manner, if the drag coefficient C_d is assumed to be a constant, equal to C_{d0}. The final expression is (check):

$$C_P = \lambda C_T + \frac{\sigma C_{d0}}{8}$$

The above expressions are true only for a linearly twisted rotor.

Ideally Twisted Rotor:

Here, the twist angle is inversely proportional to the radial location r. Such rotors are hard to manufacture, but turn out to have the lowest power consumption.

$$\theta = \frac{\theta_t R}{r}$$

Here Φ_t is the pitch angle at the blade tip.

Using this in the expression for thrust given in equation (6) we get

$$T = \frac{1}{2}\rho abc\Omega^2 \int_{r=0}^{r} \left(\theta_t \frac{R}{r} - \frac{V+v}{\Omega r}\right) r^2 dr = \frac{1}{4}\rho abc\Omega R^3 \left(\theta_t - \lambda\right)$$

Or,

$$C_T = \frac{\sigma a}{4}\left(\theta_t - \lambda\right)$$

The expression for the coefficient for power, for an ideally twisted rotor turns out to be identical to that for a linearly twisted rotor.

In summary, according to the blade element theory, the following expressions are obtained.

For a linearly twisted rotor in hover or climb,

$$\frac{C_T}{\sigma} = \frac{a}{2}\left(\frac{\theta_{.75}}{3} - \frac{\lambda}{2}\right)$$

For an ideally twisted rotor in hover or climb,

$$C_T = \frac{\sigma a}{4}\left(\theta_t - \lambda\right)$$

For both types of twist, the power coefficient is given by

$$C_P = \lambda C_T + \frac{\sigma C_{d0}}{8}$$

The first term in the power coefficient is identical to momentum theory, and is called the induced power. The second term is due to power required to turn the rotor in a viscous flow, and is called the profile power. The Figure of Merit M is given by

$$M = \frac{\lambda C_T}{\lambda C_T + \sigma C_{d0}/8}$$

DESIGN $C_{L_{max}}$:

The design $C_{L_{max}}$ for the aircraft can be calculated from the basic lift equation.

$$W = \frac{1}{2}\rho V^2 S C_L \text{ at steady state.}$$

The $C_L = \dfrac{2W}{\rho V^2 S}$

$$C_L = \frac{2(W/S)}{\rho V^2}$$

The value of wing loading is calculated from the historical data.

The density is calculated from ISA (International standard atmosphere). At an altitude of 4000m the density is found to be $0.81913 Kg/m^3$.

The velocity at cruise is 300Km/hr (83.33m/sec). Hence, the C_L for the aircraft is found to be

$$C_L = \frac{2 \times W/S}{\rho V^2}$$

$$C_L = \frac{2 \times 70 \times 9.80}{0.81913 \times 83.33^2}$$

$$C_L = 0.24145$$

AIRFOIL SELECTION:

The airfoil in many aspects is the heat of the airplane. The airfoil affects the cruise speed, take-off & landing distance, stall speed, handling qualities (especially near the stall), overall efficiency during all phases of flight.

AIRFOIL GEOMENTRY:

The front of the airfoil is defined by a landing edge radius which is tangent o the upper & lower surface.

An aircraft is designed to operate in supersonic flow will have a sharp (or) nearly-shape leading edge to prevent a drag producing bow shock.

The chord of the airfoil is the straight line from the leading edge to the trailing edge. It is very difficult to build a perfectly sharp trailing edge. So most airfoils have a blunt trailing edge with some small finite thickness.

EFFECTS OF THICKNESS:

The thickness distribution of the airfoil is the distance from the upper surface the lower surface measured perpendicular to the mean camber line, and is a function of the leading edge.

The airfoil thickness ratio (t/c) refers to the maximum thickness of the airfoil divided by its chord.

Due to fuselage effects, the root airfoil of a subsonic airfoil can be as much as 20-60% thicker than tip airfoil without greatly affecting the drag.

This is very beneficial resulting in a structural weight reduction as well as more volume for fuel & landing gear. This thicker root of the airfoil should extend to no more than about 30% of the span.

For the above condition the airfoil suited for the subsonic single seater home built aircraft as NACA23012 (5 digit airfoil).

The X and Y co-ordinate and respective plots for the selected airfoil are represented below.

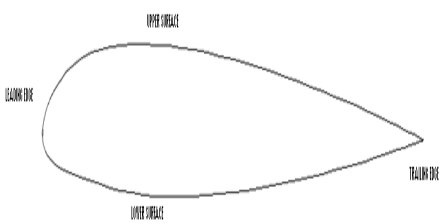
NACA23012
Lower surface

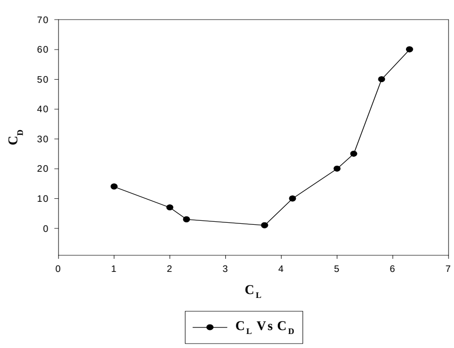
X co-ordinates	Y co-ordinates
0.999692	-0.000157
0.992377	-0.001047
0.975449	-0.003069
0.949238	-0.006100
0.914257	-0.009966
0.871190	-0.014461
0.820879	-0.019362
0.764305	-0.024443
0.702573	-0.029472
0.636886	-0.034213
0.568528	-0.038416
0.498833	-0.041814
0.429161	-0.044129
0.360874	-0.045090
0.295307	-0.044458
0.233738	-0.042059
0.177625	-0.037850
0.129534	-0.032817
0.089318	-0.028081
0.056317	-0.023623
0.030295	-0.018693
0.011638	-0.012157
0.001202	-0.028600

Upper surface

X co-ordinates	Y co-ordinates
-0.000593	0.003045
0.003555	0.016608
0.018649	0.031767
0.044889	0.46915
0.081644	0.059950
0.127322	0.069150
0.179587	0.074056
0.236342	0.075845
0.297957	0.075525
0.363488	0.073261
0.431666	0.069287
0.501167	0.063398
0.570645	0.057427
0.638751	0.050210
0.704164	0.425574
0.765614	0.034824
0.821909	0.027251
0.871954	0.020133
0.914781	0.013441
0.949556	0.008335
0.975608	0.004150
0.992431	0.001383
0.999699	0.000171

NACA23012

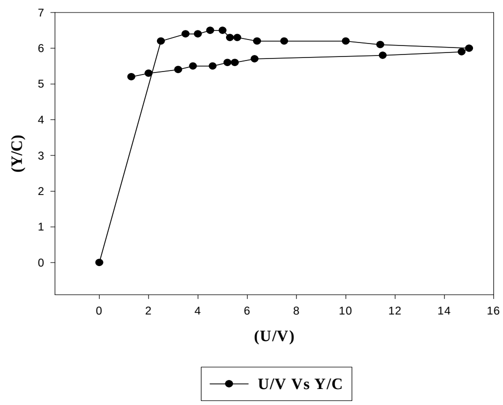


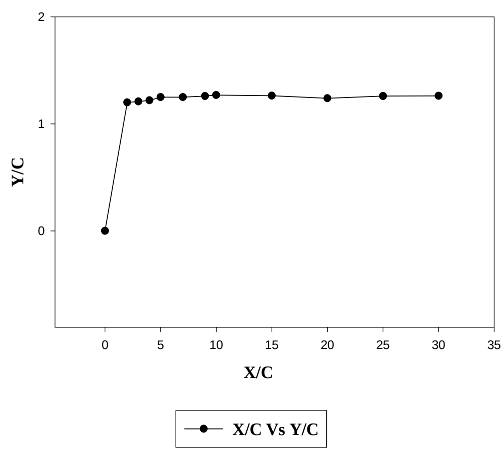


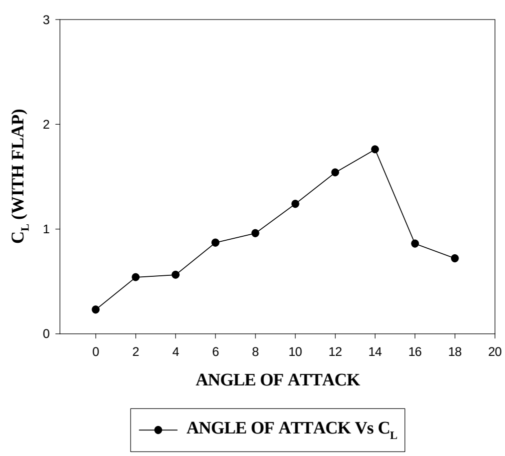
The above graph was drawn between co-efficient of lift and co-efficient of drag.

1. Negative values of C_L pertain to negative lift, which occurs when the angle of attack of the airplane is less than $\alpha_L = 0$.

2. This situation is not encountered frequently in the analysis of airplane performance, hence only that portion of the drag polar associated with positive C_L.

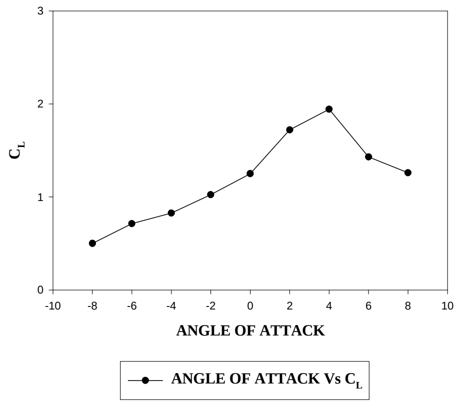






The above graph was drawn in between angle of attack and co-efficient of lift with flap.

1. Reduce the curve slope without any change of lift angle of incidence is seen in C_L Vs α characteristics.
2. Increase of stalling angle without appreciable change in maximum lift coefficient is also seen in C_L Vs α graph.
3. Definite increase in drag for every increase of lift in observed in C_L Vs C_D



The above graph was drawn between the angle of attack and the co-efficient of lift without flap.

When the plain flap is deflected, the increase in lift is due to an effective increase in camber and a virtual increase in angle of attack.

3.0 CONCLUSION

Thus the suitable airfoil was selected for single seater home built aircraft was that NACA 23012.

REFERENCES

[1] Aviation Weeks – January 2008 Edition
[2] Courtland D. Perkins & Robert E. Hage , " Airplane Performance and Stability control".
[3] Daniel p. Raymer, "Aircraft conceptual design," seventh edition.
[4] Few websites followed
[5] Ira.h. Abbott, "Theory of Wing sections"
[6] Jan Roshkam , " Airplane Design " All seven Edition.(1-7Volumes)
[7] J.D Anderson , "Aircraft Performance"
[8] John F. Fielding , " Airplane Design"
[9] L.M. Mile-Thomson, " Theoretical Aerodynamics";second edition
[10] Taylor J. Janes , "All The World Aircraft " , Janes's , London , England ,UK, 1976
[11] Thomas Cork , " Preliminary Aircraft Design"
WWW.ADL.GETCH.edu
WWW.COMBATAIRCRAFT.COM
WWW.NASA.org
WWW.Propulsion.org
WWW.ZAP16.com