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Design and Stability Analysis of a VTOL Airplane

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Abstract - *The main purpose of the project is to eliminate the relationship between runway and airplane. Due to uneven land condition in hilly terrain building or construction of a runway will be quite challenging and uneconomical. So, it is better to use an airplane which doesn't require a runway for takeoff and landings in such terrain. This can be achieved by using a VTOL i.e., Vertical takeoff and landing phenomenon. Airplane working on this phenomenon can be used to rescue soldiers from the borders during a war situation, by increasing the number of capacities of onboard load which can carry maximum people at a time without compromising speed. This type of performance is not at all possible to expect from a helicopter/chopper, because of its minimum load carrying capacity and speed. Not only for the above-mentioned application airplane working on this principle can also be used commercially for long or short distance transportation.*

This is a hybrid design it can also take off and land like a traditional aircraft by means of ground run. Our research for the project is restricted to unmanned air vehicle i.e., UAVs, so our model will just represent the phenomenon which can be further designed on a large extent. The thrust generates used in the project are nothing but the propellers driven by brushless Direct current motors, which is further controlled by a series of components which receive signal from the transmitter.

Key Words*: *UAVS, VTOL, RC Airplane, Lateral Stability Analysis, Longitudinal Stability Analysis

1.INTRODUCTION

The airplane will consist a streamline similar to other airplane the only difference will be of the positioning of the prime mover. The prime mover will generally consist an ability to rotate due to which it can deliver the thrust in different directions according to the requirement. During the takeoff the prime mover will be perpendicular to the ground to generate thrust in Y- direction which will act like a lift force. After the vehicle attain a desired altitude the angle of the rotor/prime mover will be switched to angle 'Φ'. The angle 'Φ' is decided in such manner that the vertical component of the thrust becomes equal to the weight of the vehicle. So that the vehicle can hover at a particular altitude.

And the horizontal component of thrust will be help full to give acceleration to our vehicle in X- direction.

After the vehicle attains a desired velocity in which desired amount of lift force is generated in the wings the rotor/prime mover can be further tilted to Φ=0˚ which will be parallel to the land. Due to this, the amount of thrust generated by the prime mover can be focused on the acceleration of the vehicle.

2. Design

Let us consider our airplane to be of 2Kg (assumed) and the thrust generated by the propeller be 4kg (assumed). After the motor tilts we see that, we have 2 components of force obtained, horizontal component which will help in the acceleration and vertical component which will act as a lift force, the angle of the tilt is decided in such a manner that the vertical component lift will be equal to weight of the vehicle so that the stalling of the vehicle is avoided.

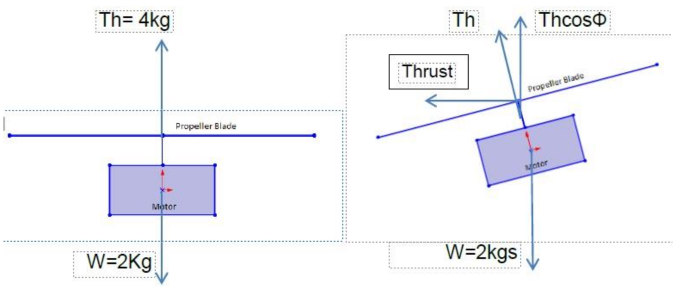


Fig -1: Propeller Orientation

$F_L = \text{Cos}\Phi$

$2x9.81 = 4x9.81\text{Cos}\Phi$

$\Phi = 60^0$

On the other hand, we also calculate the horizontal thrust force T_{HX},

$T_{Hx} = T_H\, \text{Sin}\Phi$

$T_{Hx} = 4x9.81\text{Sin}\,(60)$

$T_{Hx} = 34\text{ N}$

Hence, we see that, 34N of horizontal thrust force will be responsible for accelerating the vehicle, let us apply De-Alembert's principle to calculate the acceleration

$\sum F_x = mxa$

m- Mass in Kg

a- Acceleration in m/s

Let us assume overall drag force to be 1 kg (Assumed)

34-9.81=2xa

a=12.545 m/s

Let us assume the initial velocity of the airplane to be 0 and distance covered under the application of tilt be 50m.

Applying Kinematics (constant acceleration)

S=50 m

a=12.545 m/s

u=0

S=ut+ a

50=0+ x12.545x

t= 2.823 secs

Now let us calculate the velocity of the vehicle after the displacement of 50 m in 2.82 secs

V=u+at

V=0+12.545x2.823

V=35.414 m/s

But the airplane we designed can generate the required amount of lift at 20 m/s, so after the attainment of 20 m/s the thrust generators can be further tilted to 0^0 which will be parallel to the ground so that the thrust can be further concentrated towards the acceleration of the vehicle.

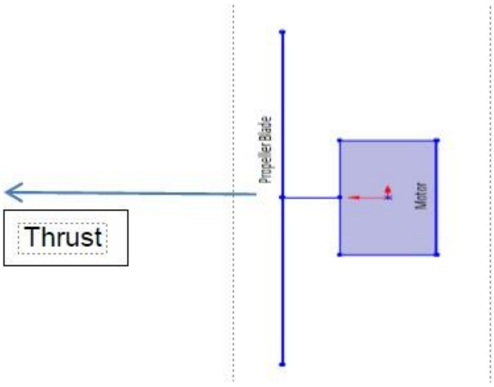


Fig -2: Thrust orientation

The above-mentioned data are assumed and can be different for the actual model accordingly but the approach will remain the same.

3. Design of Wings

Before the start of the design, it was assumed that the overall weight of the vehicle including the payload will be 2kgs.

So now to lift the whole airplane we needed our primary lifting surface i.e., our wings, so we firstly design the wings.

We have considered or assumed the maximum speed of our airplane to be 50m/s, for the calculation of Mach number.

$$\text{Mach.no} = \frac{Velocity}{Speed\ of\ sound}$$

Mach.no = 0.1457

As the Mach Number comes out to be subsonic four-digit airfoil can be used for the application.

The Lift equation=$0.5\rho v^2 s C_L$ (Derived from Bernoulli's equation)

ρ- Density of fluid i.e air, ρ=1.2 kg/m^3

v- Velocity of the vehicle in m/s

s- Lifting Area in m^2

C_L- Coefficient of Lift

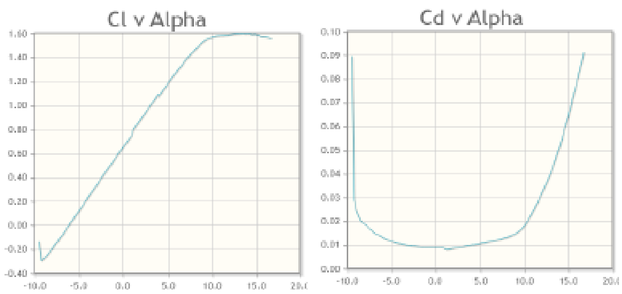
Desired lift force generation by the wings will be equal to the weight of the vehicle i.e., 2kgf or 19.62N at a speed of 20m/s and also assuming lifting area to be 0.13m^2(by rough estimation)

$19.62=0.5*1.22*20^{0.2}*0.13*C_L$

C_L=0.62

Hence, from the above value of it is required to select an NACA (National Advisory Committee for Aeronautics) airfoil which is suitable for the application.

Selected NACA-6412 airfoil for the given application for the calculated value of Reynolds number.



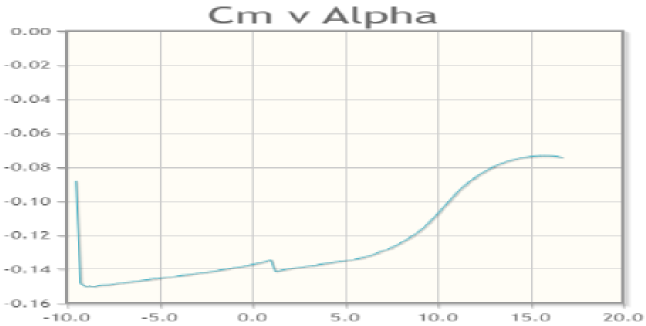


Fig-3: Data for NACA-6412

The data in the form of graph displayed above for the airfoil shape NACA-6412 will be used in the design further.

We see that the value for NACA-6412 is 0.62 at an angle of attack of 0^0

$19.62=0.5*1.22*20^2*s*0.62$

$S=0.13m^3$

$0.13m^2$ of lifting surface area is required to generate the lift force of 19.62N at 20m/s of velocity. Let us now calculate the dimension of wing, The dimension of the wing can be calculated with the help of ASPECT RATIO (A.R) and the aspect ratio for the application i.e., Average cruising flight, is 5.6.

$AR=span^2/Lifting force$

$5.6=Span/0.13$

Span=0.866 m

Total Lifting Area=Span*Chord

0.13=0.8866*Chord

Chord=0.1496 m

Chord = 0.1496m Let us now Calculate the wing-cub loading (C) of the wing. Wing-cub loading ranges 1-7 (Floater)

8-12 (Good Flyer)

13 (Will demand high velocity to generate desired Lift)

$C=Weight/Lifting Area^{0.5}$

$10.5=2000/ 13^{0.5}=42.86$

Weight in Grams & Area in

For a good flyer wing-cub loading should be between 8 to 12,

If above calculated value is accepted than the airplane or wings will be demanding very high velocity to generate the required amount of lift and eventually to generate high velocity high thrust will be required which will on other hand demand high power, so accepting this value will not be efficient.

Re-Designing,

Let us take the wing-cub loading to be as 10.5 to bring it under good flyer range.

Now Lifting area $=33.105 dm^2=0.331 m^2$

Let us now calculate the Span and Chord of the wing by using the Aspect Ratio

$AR=span^2/Lifting Area.$

$5.6=span^2/0.331$

Span=1.36 m

Lifting Area= Span*Chord

Chord=0.243 m

Fig-5: Wing Span

Let us calculate the maximum velocity on which the desired lift will be generated.

$16.92=0.5*1.22*V^2*0.331*0.021$

V=13.82 m/s

We see that, 13.82 m/s of speed is required to generate the lift force which will be equal to the weight of the vehicle

Now let us calculate the drag generation only for wing alone configuration, According to NACA-6412 the drag coefficient (CD) at 0^0 angle of attack is 0.021.

$F_D=0.5*P*V^2*S*C_D$

$F_d=0.809N$

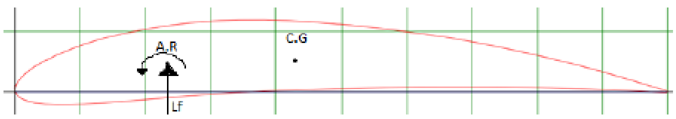


Fig-4: Wing Profile NACA 6412

We know that the lift force generated by the wings is concentrated at the center of pressure.

But it is also known that at the Aerodynamic center (A.R) moment generation is constant at a particular angle of attack irrespective of the speed of the vehicle

This moment is due to shifting the concentrated force from the center of pressure to the Aerodynamic center of the wing and which is about its C.G for wing alone configuration.

So let us calculate the nature and magnitude of this moment and the behavior of the wing.

If the moment generated is nose down then the wing is stable, for a cambered airfoil the Aerodynamic center is always ahead of its C.G due to this, always nose down effect is witnessed in cambered airfoil, Aerodynamic Center is locating at a distance of Chord/4 from its leading edge for cambered airfoil and Chord/2 for symmetric airfoil.

According to NACA-6412 at 0^0 Angle of attack Coefficient of Moment (C_M) is given as -0.135, Negative value of indicates nose down moment

The Moment Equation

Moment=$0.5PV^2SC_MX$

C_M-Coefficient of Moment

X-Mean Chord

Moment = -1.2931 Nm (nose down moment)

This Moment Generated is a nose down moment and it is absolutely undesirable which is disturbing the stability of the wings which will result in instability of the whole airplane

Hence, to eliminate this moment to make the airplane stable a counter moment is required of same magnitude but off opposite sense i.e., nose up moment.

So, to counter the induced moment Horizontal Stabilizer (H.S) is used.

4. Fuselage Design

Finess ratio (F.R) is decided to be 12 for an average cruising flight, for the accommodation of payload i.e., Electronics components the width inside the fuselage is required to be of at least 100mm

Finess ratio is given as,

FR=Length of Fuselage/Internal Diameter

12= Length of Fuselage/97

Length of Fuselage=1160mm

The Cross section of the fuselage decided to select from the NACA four-digit series and NACA-0010 symmetric airfoil is selected for the fuselage cross section.

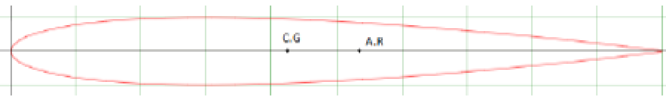


Fig -5: Fuselage Profile NACA-0010

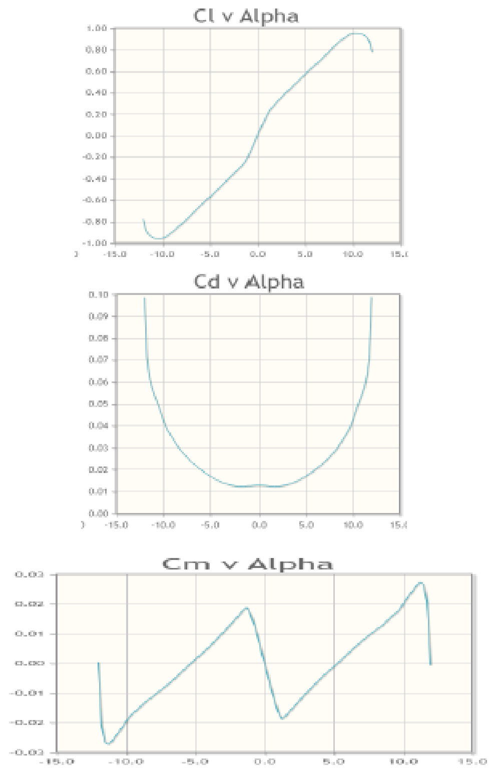


Fig-7: Data on NACA 0010

Let us check the behavior of fuselage alone configuration

For a symmetric airfoil at 0^0 Angle of attack there will no lift generation, so due to this no moment will be induced, but there will e some drag due to the wind blast, so we see that at 0^0 angle attack fuselage contribution is almost negligible.

Calculating the drag generation at $\alpha=0^0$,

$F_D=0.5*P*V^2*S*C_D$

C_D=Coefficient of Drag

$F_D=0.5*1.22*133.82*0.122*(-0.012)*1.160$

$F_D=0.2N$

Let us check the longitudinal stability of the fuselage alone configuration at $\alpha=2.5^0$,

Let us check the longitudinal stability of the fuselage alone configuration at $\alpha=2.5^0$,

According to NACA-0010 C_M at $\alpha=2.5^0$ is given as -0.0120 since the value is negative it will generate a nose down moment

Let us calculate the magnitude,

Moment=0.5*P*V^2*S*C_M*X

Moment=0.5*1.22*13.82^2*0.112*(-0.0120) *1.160

Moment=-0.181Nm (Nose down)

Hence, we see that the fuselage is generating a nose down moment at positive α, hence it consists an ability to come back to its initial cruising stage therefore, it can be concluded that the airplane is statically stable

But according to the data of NACA-6412 the fuselage will start producing a nose up moment after the α reaches more than 5^0, the limit to pitch the airplane can be commented when the stability analysis is performed by considering all components of the airplane.

But it is essential to trip the airplane at α=0^0 so that smooth cruising is obtained, for trimming the airplane there is a need of horizontal stabilizer which can eliminate the nose down moment and trip the airplane at α=0^0.

5. Design of Horizontal Stabilizer

Before designing the horizontal stabilizer, it is important to understand the behavior of the airplane without the horizontal stabilizer.

The wing is generating a nose down moment at α=0^0 so it is required to place the stabilizer in such a way that will create a same magnitude of moment but in opposite direction.

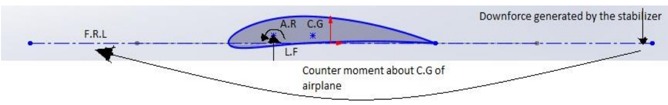


Fig-8: Horizontal Stabilizer orientation

The wings are placed in such a way that the A.R of the wing is made coincident with the C.G so that there is no further moment generation due to eccentricity of the lift force concentrated point and the C.G of whole airplane which can make the airplane unstable.

Horizontal stabilizer is placed somewhere behind the wing near the trailing edge of the fuselage, so it is necessary to calculate the distance between the A.R of the horizontal stabilizer and the C.G of whole airplane, and this will totally depend upon what amount of downforce to be generated by the horizontal stabilizer to trim the airplane at α=0^0.

6. Calculating the required downforce required,

Let us assume the distance between the C.G of airplane and A.R of Horizontal stabilizer to be 300mm

Moment to generate= Downforce x Lever arm

-1.2931= Downforce x 0.3

Downforce= -4.625N

Required down force = - 4.625 N

F_L = 0.5ρ*v^2*S*$C_{L\alpha}$

-4.625=0.5*1.22*1.382^2*S*(-0.3)

S=0.1323

Let us now calculate the dimension of the horizontal stabilizer using the Aspect ratio, for an average cruising flight aspect ratio for the stabilizer is considered as 4

FR=span/area

4=Span/0.1323m

Span=0.7274m

Chord=0.1818

Calculating the wing-cub loading for horizontal stabilizer,

C=W/$S^{1.5}$

C=471.5/$13.23^{1.5}$=9.97

We see that, the value of the wing-cub loading for the horizontal stabilizer comes in the optimal range hence the assumptions made before designing the horizontal stabilizer turns out to be correct, but from proper respond between the wings and the horizontal stabilizer both wing-cub loading magnitude should be same hence, changing the wing cub-loading to 10.5 to minimize further losses.

Redesigning,

C=10.5 (Similar Wing)

10.5=471.5/$S^{1.5}$

S=0.12633m

Calculating the new span and chord by using the aspect ratio which is „**4**" for the horizontal stabilizer,

R=Span/Area

4=Span/0.12633

Span=0.71m

Chord=0.177m

F_L-4.415

For the above application it is decided to select NACA-0012 symmetric airfoil with a negative setting angle which will be, α=-15°.

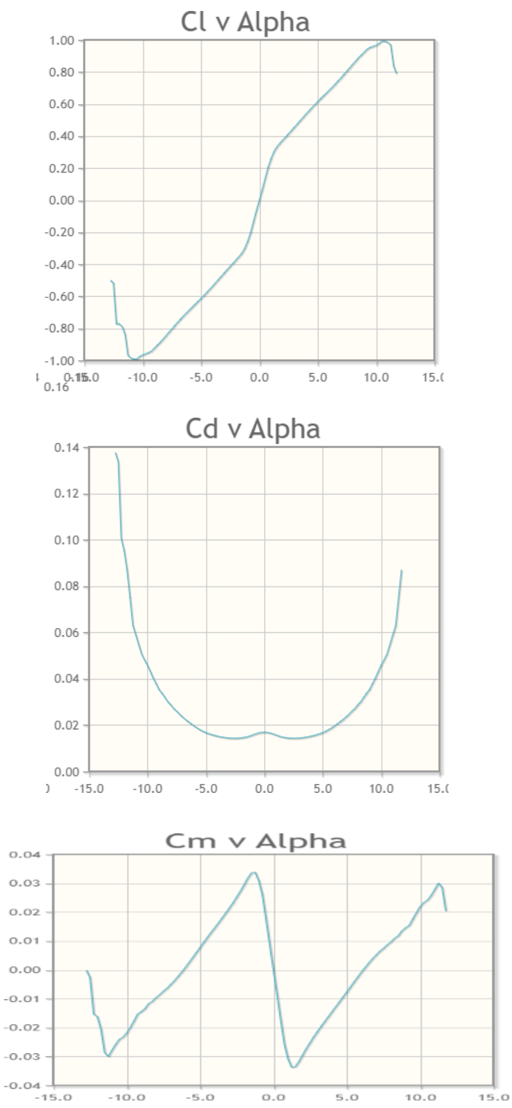


Fig-9: Data for NACA0012

Calculating the drag force generation,

F_D=0.5*P*V*S*C_D

F_D=0.5*1.22*13.82²*0.1263*0.018

F_D=0.2648N

6. Calculating the stability for horizontal stabilizer alone configuration.

From the C_M vs α graph of NACA-0012 we see that for α=-1.5^0 is given as „0.016" which is positive which means it will generate a nose up moment.

Let us calculate the magnitude of this moment

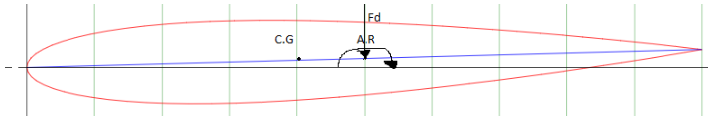


Fig-10: Profile of NACA 0012

Moment=0.5*P*V²*S*C_M*X

Moment=0.5*1.22*13.82²*0.1263*0.1263 *0.178

Moment=0.04191N (Nose Up)

7. Moment=0.5*1.22*13.82²*0.112*(-0.0120) *1.160

To make the airplane stable for cruising it is necessary to trim the airplane at α=0^0 Therefore, applying the dynamic

equilibrium condition,

∑M@ CG of airplane=0 (dynamic equilibrium condition)

-M. W + M. HS + stabilizer × Xcg= 0

-1.2931 + 0.0491 + 4.415 × Xcg= 0

Xcg= 0.2834

We see that the total length of fuselage is 1160mm, hence the distance from the cg to aerodynamic center of stabilizer will be 0.588.

-1.2931 + 0.04191 + × 0.580 = 0

Required down force = -2.1572 N

So, we see, that due to the extension or increment of lever arm the downforce requirement reduces hence, now we have two methods to redesign the horizontal stabilizer either we can change the angle of attack or we can reduce the span and chord of the stabilizer, it is better and efficient to reduce the dimension rather than changing the angle of attack because reduction of dimension will also cause in reduction of weight which will be more efficient.

F_D= 0.5ρ*v^2*S*C_L

-2.1572=0.5*1.22*13.82²*S*-0.3

S=0.06171

4=Span²/area

Span=0.4968

Chord=0.1241

Fig-11: Chode

Now, after trimming the airplane at we see that, the dimensions of the fuselage wings and also the horizontal stabilizer are quite huge according to the requirement, so what we decided is to scale or design at ratio of 1:2.

8. Dimensions after scaling in millimeter's (mm) Scale ratio (1:2)

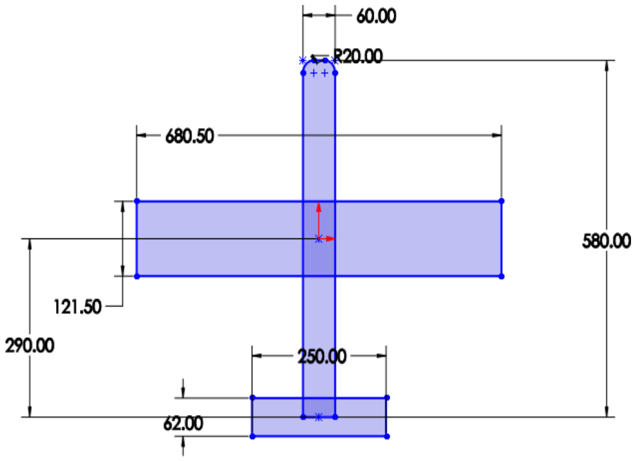


Fig-12: Final Scaled Model

9. Reassuring the stability after scaling the design

As we have scaled the whole design it is essential to check the stability and other parameters to make sure that the design is on safer side

Let us first calculate the lift and drag generation of the wings after scaling,

$Lift_W = \times \rho \times v^2 \times S \times C_{L0}$

$Lift_W = 0.5 \times 1.2 \times 20^2 \times 0.08268 \times 0.6$

$= 11.905$ N or 1.21 kgf

$Drag_W = \times \rho \times v^2 \times S \times$

$Drag_W = 0.5 \times 1.2 \times 20^2 \times 0.08268 \times 0.021$

$= 0.4167$ N

Let us calculate the change in nose down moment of the wing,

$Moment = 0.5 \times \rho \times v^2 \times S \times C_{M0} \times C''$

$= \times 1.2 \times 20^2 \times 0.08268 \times (-0.139) \times 0.1215$

$= -0.3351$ Nm (Nose down moment)

Let us now calculate the change in downforce, drag force and moment in Horizontal stabilizer,

Let us now calculate the change in downforce, drag force and moment in Horizontal stabilizer,

$Downforce_S = 0.5 \times \rho \times v^2 \times S \times C_L$

$Downforce_S = 0.5 \times 1.22 \times 20^2 \times 0.01542 \times 0.4$

$Downforce_S = 1.4803$ N (↓)

$Drag_S = 0.5 \times \rho \times v^2 \times S \times C_{D2.5}$

$Drag_S = 0.5 \times 1.2 \times 20^2 \times 0.01542 \times 0.022$

$Drag_S = 0.0740$ N

$Moment_S = 0.5 \times \rho \times v^2 \times S \times C_{MO} \times C''$

$Moment_S = \times 1.2 \times 20^2 \times 0.01542 \times 0.025 \times 0.002$

$Moment_S = 0.0053$ Nm.

There will be no lift and moment for fuselage at $\alpha=0^0$, except of the drag force

$Drag_S = 0.5 \times \rho \times v^2 \times S \times C_D$

$Drag_S = 0.5 \times 1.2 \times 20^2 x 0.0348 x 0.0125$

$Drag_S = 0.1044$ N

10. Checking the stability at Î±=0â° Including Fuselage and drag effect.

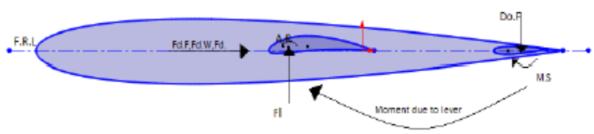


Fig-13: checking stability at α=0

Applying the condition of dynamic equilibrium,

∑M@ C. G= 0 Counter Clockwise +ve

$-Momen_{Wing} + Moment_{Stablizer} + Do.F * Xcg = 0$

$-0.3351 + 0.00593 + 1.4803 \times Xcg = 0$

$Xcg = 0.2226$m

Since, the calculated value of lever arm is 0.29m let us not change the position of the horizontal stabilizer instead we

change its setting angle to generate required amount of force.

∑M@ C. G= 0 Counter Clockwise +ve

$-Momen_{Wing} + Moment_{Stablizer} + Do.F*Xcg = 0$

NACA 6412	NACA 0012	NACA 0010
C_{M5} =-0.14	C_{M5}= -0.025	C_{M5} = 0
C_{L5}= 1.15	C_{L5}= 0.4	C_{L5} = 0.6
C_{D5} = 0.025	C_{D5} = 0.03	C_{D5} = 0.018

$-0.3351 + 0.00593 + Do.\,F \times 0.29 = 0$

Downforce = -1.135 N

$Fd = \times \rho \times v^2 \times S \times C_L$

$-1.135 = 0.5 \times 1.22*13.82^2*0.01542*C_L$

C_L-0.301

According to NACA-0012 it will serve the application at a setting angle of $\alpha = -1.5^0$ Checking the stability at $\alpha=1.5^0$

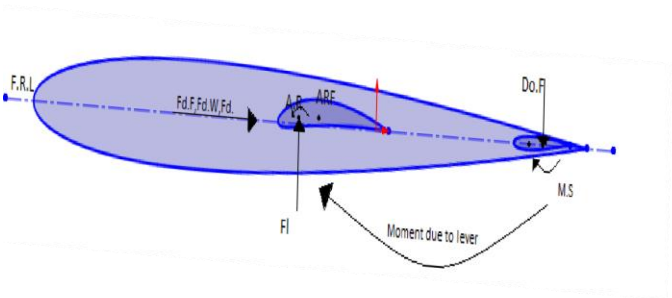


Fig-14

At an $\alpha=1.5^0$, we noticed that the horizontal stabilizer will be at an angle of attack of 0^0 i.e., $\alpha=0^0$, that means it will not contribute in generating a nose moment, since there will be no lift and moment generation, so now at this stage the fuselage will take care to produce a nose down moment and to make the airplane trip at $\alpha=0^0$, the dynamic equilibrium equation for this condition is given below

$L_W = 16.86$ $L_S = 0$ $M_W = -0.335Nm$

$D_W = 0.4167$ $D_S = 0$ $M_S = 0$

$L_F = 3.2256Nm$ $M_F = -0.00903Nm$

$D_F = 0.1290$

The above values for forces and moments for various component like wing, fuselage and horizontal stabilizer are calculated according to their data in their respective Cl vs α, Cd vs α and Cm vs α graphs which is stated above earlier.

∑M=0 +ve

$-0.335 - 0.00903 - 3.2256 \times (Xcg - ARF) \neq 0$

We see that the above equation is not coming out to be ‘0’ and will have some magnitude in a nose down moment sense hence, we conclude that airplane is trying to come back to its trimmed position therefore, we can state that airplane is statically stable.

Checking the stability at $\alpha=5^0$

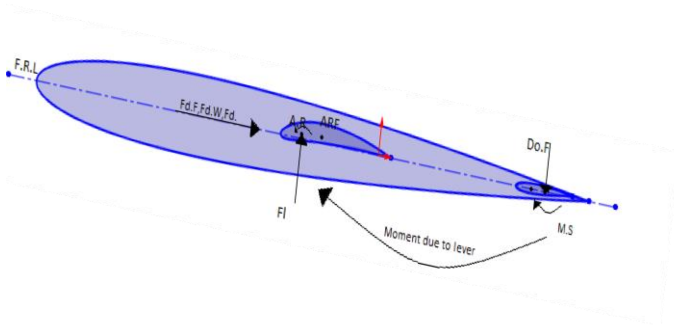


Fig-15: Checking Stability at α=5°

∑M=0 +ve

$-0.14 - 0.025 - \times (Xcg - ARF) - \times Xcg \neq 0$

We see that from the dynamic equilibrium equation at an angle of attack of 5^0 the fuselage and horizontal stabilizer both contributing to trim the airplane at 0^0 hence, it can be stated that the airplane is statically stable.

From above Complete stability analysis we can jump to a conclusion that the design carried out is good and the airplane is performing the following maneuver’s very smoothly without any difficulty

The CAD model for the design is given below,

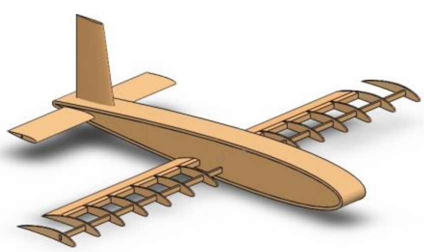


Fig-16: 3D Model

11. CONCLUSION

The above design and analysis found to be correct after performing the stability analysis run by using finite element method (FEA), the positive result of the herewith analysis carried out be finite element method give us green signal to go ahead for the fabrication of the model as all the design and estimations are found to be correct at the end.

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