Conceptual Design and Performance Estimation of a 15-Seater Propeller Driven Private Airplane

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Abstract: Propeller is a mechanical device consist of hub and blades which produce thrust due to the rotational motion of blades mounted on the hub. In this study a low speed propeller driven private aircraft is designed which carries passenger with luggage and crew member. Geometric analysis is done for twisted wing configuration where two different airfoil NACA 23012 and NACA 23015 at tip of the wing and root of the wing are applied respectively. ANSYS is used for the creation of geometry and meshing and FLUENT is used as a solver to numerically analysis the airfoils of wing configuration. The performance parameters of the airplane is estimated as per the requirements.

Keywords: Airplane Design, Low speed airplane, NACA 23012, NACA 23015, Propeller driven Aircraft

Nomenclature

AR = Aspect ratio
α = Angle of attack
C_{D_0} = coefficient of drag at zero lift
C_{Lmax} = Maximum coefficient of lift
$\frac{L}{D}$ = Lift to drag ratio
P_R = Power required during takeoff
S_g = Ground roll
S_l = Landing distance
V_{stall} = Stalling velocity of the airplane
$\frac{T}{W}$ = Thrust to weight ratio
$\frac{W}{S}$ = Wing loading
W_0 = Gross weight of airplane during takeoff
W_{crew} = Weight of crew members
$W_{Payload}$ = Weight of the passengers and luggage
W_{Fuel} = Weight of fuel
W_{Empty} = Structural and instrumental weight of airplane or empty weight of the aircraft
θ_{OB} = Flight path angle to clear obstacle

I. INTRODUCTION

Design of an airplane starts with set of specifications and new innovative ideas. Before manufacturing of an airplane there are three phase of design. First phase is called as conceptual design, second phase preliminary design and the last one is detail design. In conceptual design phase the configuration of the airplane is considered as per the requirements. The shape, size, weight and the performance parameters are estimated during the conceptual design phase. The most important parts of the conceptual design deals with the aerodynamic, propulsion system and flight mechanics of the airplane.

In this paper we are concerned about the conceptual design of a low speed propeller driven aircraft carrying 12 passengers with luggage and 3 crew members, flying at maximum level speed of 280 mph or 450 kmph at mid cruise and range of the aircraft is 2800 km. The aircraft is climbing at a speed 1200 ft/min and service ceiling is 28000 ft. The stalling velocity of the aircraft is 70 mph or 113 kmph and the landing distance and takeoff distance is respectively is 2200ft and 2500 ft to clear an obstacle of 50ft high. A geometrically twisted wing configuration is used where two different airfoils NACA 23012 and NACA 23015 respectively at tip and root of the wing.

II. WEIGHT ESTIMATION OF THE AIRPLANE

Considering the entire gross weight of airplane is W_0 during takeoff, which is mentioned bellow:

$$W_0=W_{crew}+W_{Payload}+W_{Fuel}+W_{Empty} \quad (1.1)$$

$$\text{Or, } W_0=\frac{W_{Crew}+W_{Payload}}{1-\frac{W_{Fuel}}{W_0}-\frac{W_{Empty}}{W_0}} \quad (1.2)$$

Assuming $\frac{W_{Empty}}{W_0}$=0.62 from the historical data of different aircraft and to estimate $\frac{W_{Fuel}}{W_0}$ the amount of fuel required to complete mission depends on the efficiency of the propeller efficiency, propulsion device, the engine specific fuel consumption and also depends on lift to drag ratio. From the Brequet range equation,

$$R=\frac{\eta_{pr}}{C}\frac{L}{D}\ln\frac{W_0}{W_1} \quad (1.3)$$

Mission segment weight fraction can be written as $\frac{W_i}{W_{i-1}}$ .So the ratio of the weight at the end of the mission to the initial weight of the aircraft can be written as

$$\frac{W_5}{W_0}=\frac{W_1}{W_0}\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4} \quad (1.4)$$

The variation in weight during each segment is due to the consumption of fuel. Assuming at the end of mission the tank is empty. So we can write W_{Fuel} = 1- $\frac{W_5}{W_0}$. However at the end of the mission the fuel tanks are not completely empty. There is 6% allowance to reserve and trapped fuel.

$$\text{So, } \frac{W_{Fuel}}{W_0}=1.06\left(1-\frac{W_5}{W_0}\right) \quad (1.5)$$

During the takeoff segment the $\frac{W_1}{W_0}$ is considered as 0.97 and during the climb segment $\frac{W_2}{W_1}$ as 0.985.For cruise segment using the Brequet range equation the value of $\frac{W_3}{W_2}$ can be obtained. Assuming $(\frac{L}{D})_{max}=14$ and specific fuel consumption for this aircraft engine is 2.02×10^{-7} sec^2/ft and the propeller efficiency of a variable pitch propeller is 0.85.

$$2800\times 5280=\frac{0.85}{2.02\times10-7}\times 14\times \ln\frac{W_2}{W_3}$$

$$\text{Or, } \frac{W_3}{W_2}=0.898$$

During descending segment $\frac{W_4}{W_3}$ = 1 and landing segment of the mission $\frac{W_5}{W_4}$ = 0.995.

So, after calculation we can find out $\frac{W_5}{W_0}$ = 0.8537 and $\frac{W_{Fuel}}{W_0}$ = 0.1551.

Now consider the average weight of the passenger as 180 lb and baggage per person as 40 lb in cargo hold and the average weight of the crew as 180 lb.

Total W_{crew} = 180×3 =540 lb

And $W_{Payload}$ = (180×12) + (40×12) = 2640 lb

$$W_0=\frac{540+2640}{1-0.1551-0.62}=14140 \text{ lb}$$

Finally we can calculate the fuel weight using the above mentioned gross weight of the aircraft and the fuel weight is

$$W_{Fuel}=W_0\times\frac{W_{Fuel}}{W_0}$$

$$=14140\times 0.1551=2194 \text{ lb}$$

As we know the weight of the aviation gasoline is 5.64 lb / gal.

Capacity of fuel tank = $\frac{2194}{5.64}\approx 390$ gal

III. ESTIMATION OF C_{Lmax} OF AIRPLANE

In this aircraft we are using different airfoil at different position of the wing. The aircraft has a geometrically twisted wing in which NACA 23012 airfoil is using at the tip of the wing and NACA 23015 is using at the root of the wing. In both of the airfoil the maximum chamber is placed at 15% of the chord length and the thickness are 12% and 15% respectively. Using of these different thickness of airfoil produce structural strength to the wing and aerodynamically the NACA 23015 stalls at a small angle of attack than NACA 23012 airfoil. From the numerical analysis of both of these airfoils using Ansys workbench we get the $C_{L_{max}}$ as 1.5 for NACA 23015 and 1.4 for NACA 23012 .

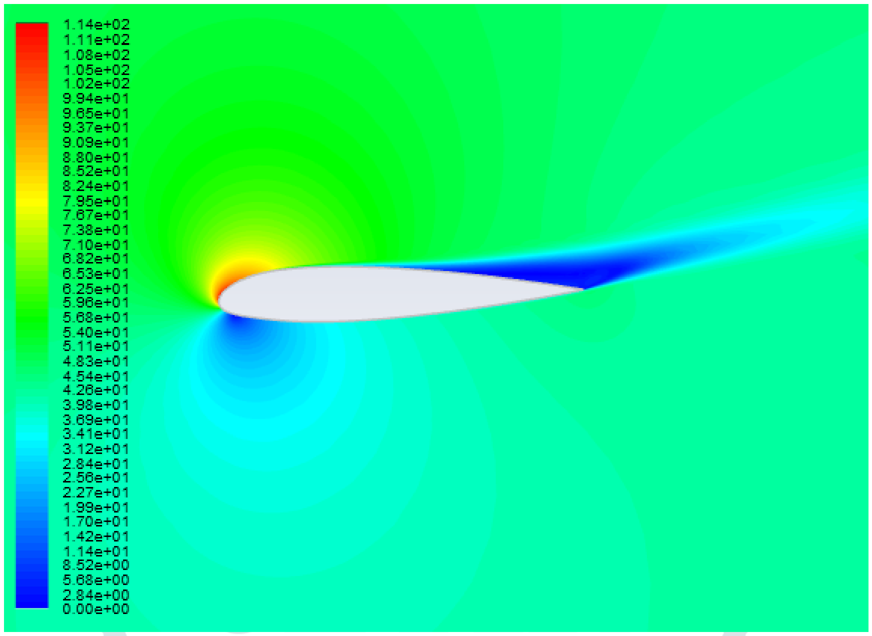

Fig 1: Velocity contour of NACA 23015 at $C_{L_{max}}$

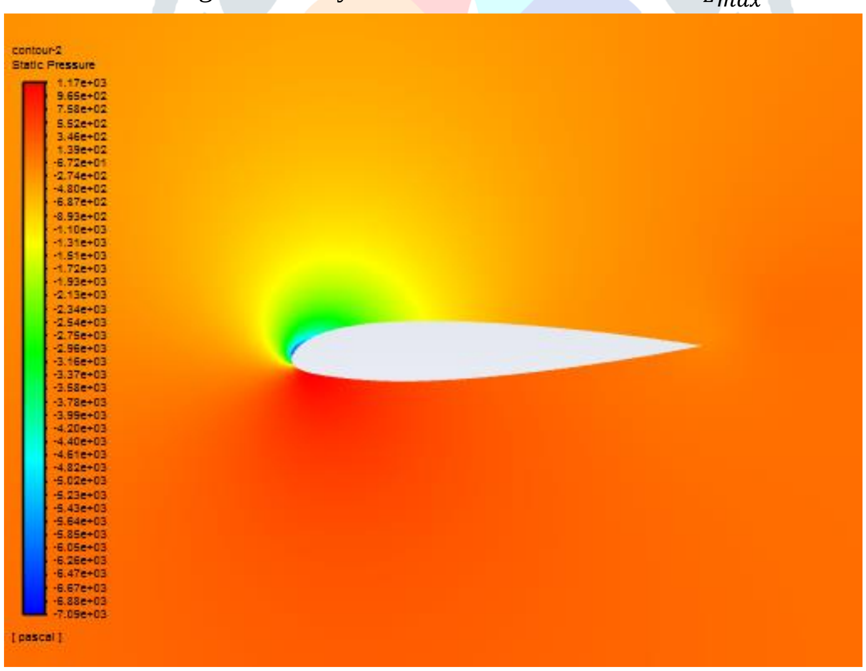

Fig 2: Pressure contour of NACA 23015 at $C_{L_{max}}$

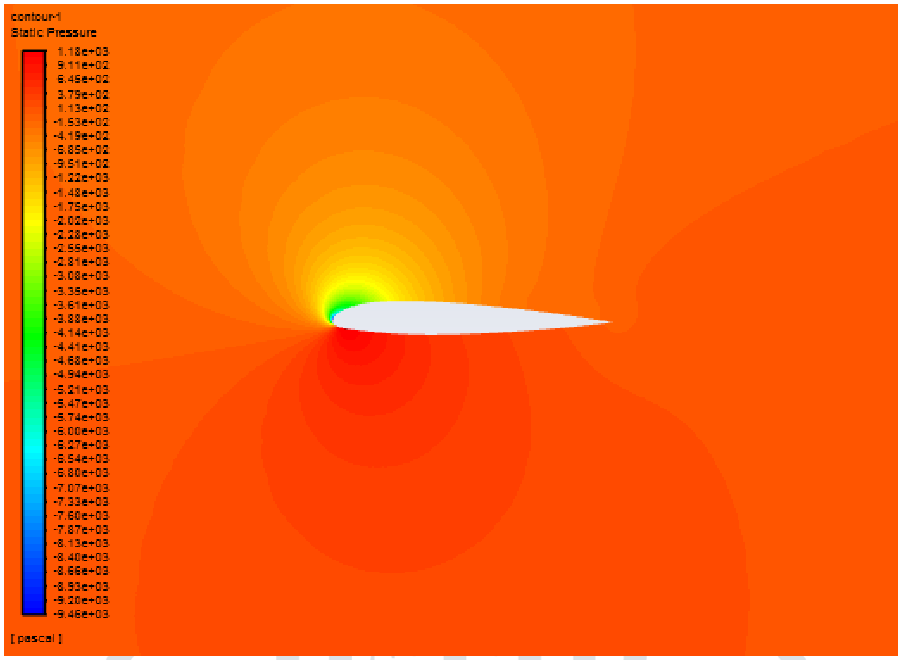

Fig 3: Velocity contour of NACA 23012 at $C_{L_{max}}$

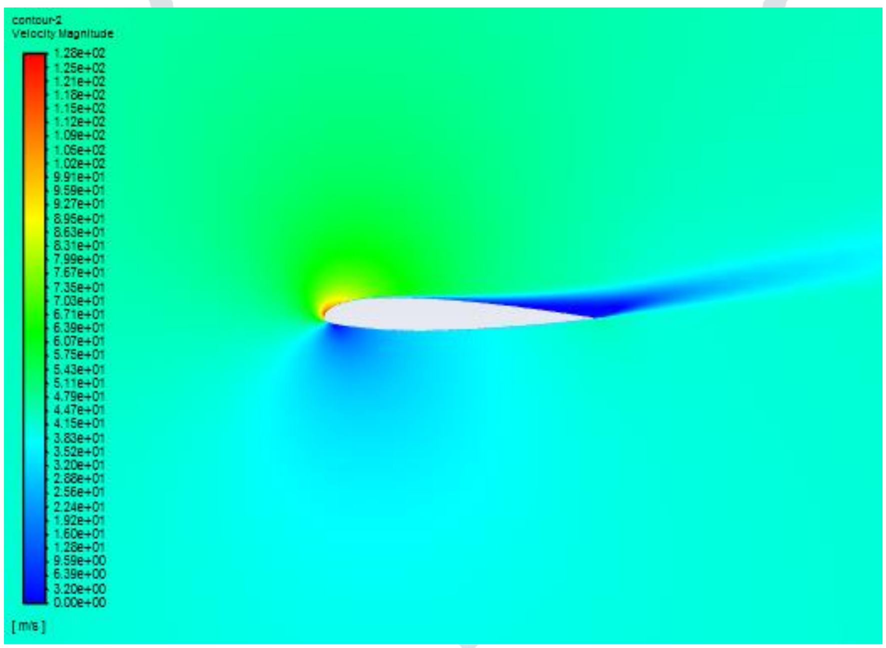

Fig 4: Pressure contour of NACA 23012 at $C_{L_{max}}$

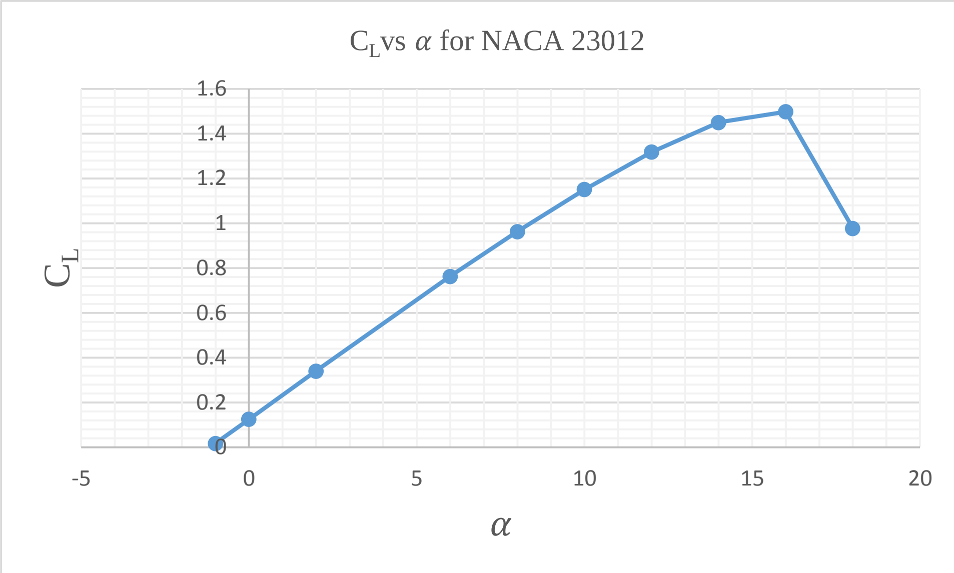

Fig 5: Coefficient of lift vs Angle of attack of NACA 23012

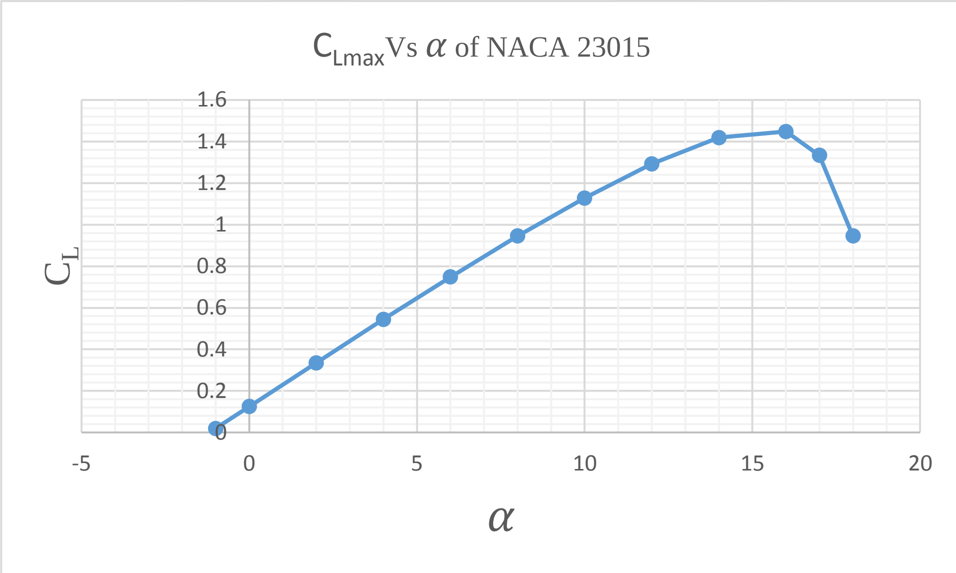

Fig 6: Coefficient of lift vs Angle of attack OF NACA 23015

Taking the average from both of the airfoil $(c_l)_{max}$ as 1.45 and we designing a wing with flap at trailing edge. From the historical data from different aircraft design we consider a flap with 45° deflection increase the $(c_l)_{max}$ 0.9 so the average $(c_l)_{max}$ becomes (1.45+0.9) = 2.35. According to Raymer for finite wing design with an AR greater than 5 has $C_{L_{max}}$= 0.9×$(c_l)_{max}$ = 2.115. Here we are neglecting the effect of fuselage, tail and other parts of aircraft.

IV. ESTIMATION OF WING LOADING

Wing loading of an airplane plays an important role during aircraft design. Landing distance and takeoff distance depends upon the wing loading of the aircraft and wing loading also plays a vital role with the maximum speed of the aircraft as V_{max} increase the Wing loading will also increase. As our airplane is a low speed propeller driven aircraft we are concerned about V_{stall} and it can be expressed as

$$V_{stall} = \sqrt{\frac{2}{\rho_\infty}\frac{W}{S}\frac{1}{C_{Lmax}}} \qquad (2.1)$$

Using (2.1) and substitute the stalling velocity as 70 mph or 102.7 ft/sec and ρ_∞ = 0.002377 slug/ft^3 at sea level we got the value of wing loading as $\frac{W}{S}$ = 26.5 lb/ft^3.

Now using the landing distance we can calculate the $\frac{W}{S}$ for better accuracy. Landing distance (S_l) is the summation of the approach distance (S_a), flare distance (S_f) and the ground roll (S_g) of the aircraft.

$$S_l = S_a + S_f + S_g \quad (2.2)$$

From the historical data from different airplane we came to know that approach angle $\theta_a \leq 3^o$. Using the formula of flight path radius, R = $\frac{{V_f}^2}{0.2g}$ (2.3)

And V_f = 1.23× V_{stall} = 126.32 ft/sec and we got R = 2478 ft and flare height (h_f) = R(1-cos θ_a) = 3.4 ft.

To clear obstacle of 50ft is $S_a = \frac{50-h_f}{Tan\theta_a}$ = 890 ft and flare distance S_f = RTanθ_a = 130ft. So the ground roll is 1180 ft and ground roll can be expressed as $S_g = jN\sqrt{\frac{2}{\rho_\infty}\frac{W}{S}\frac{1}{C_{Lmax}}} + \frac{j^2(W/S)}{g\rho_\infty C_{Lmax}\mu_r}$ (2.4)

For a commercial airplane j = 1.15, N=3, and μ = 0.4, using (3.4) and solving the quadratic equation we get the ground roll as 37.21lb/ft^2. Comparing different values of $\frac{W}{S}$ it is clear that if $\frac{W}{S}$ < 37.21 lb/ft^2 then the landing distance is less than 2200ft which is satisfying the requirements. Using the $\frac{W}{S}$ value we get the wing area as 533.6 ft^2.

V. ESTIMATION OF $\frac{T}{W}$ AND P_R of AIRPLANE

Thrust to weight ratio of an airplane plays a vital role to determine takeoff distance, rate of climb and maximum velocity. Here as per requirements of aircraft the takeoff distance is 2500 ft to clear a 50 ft obstacle. Using the equation of ground roll

$$S_g = \frac{1.21(W/S)}{g\rho_\infty C_{Lmax}(T/W)} \quad (3.1)$$

Assuming a flap deflection of 20 ° during takeoff will increase the Coefficient of lift 0.5 and it will become 1.95. Using (3.1) we get Stall velocity corresponding to the $C_{L_{max}}$ = 1.95 as V_{stall} = 106.93 ft /sec. Flight path radius of an airplane can be expressed as

$$R = \frac{6.96\times Vstall^2}{g} \quad (3.2)$$

Using (3.2) we get range as 2472 ft and determine $\theta_{OB} = Cos^{-1}(1 - \frac{h_{OB}}{R})$ = 11.54° and airborne distance S_a = RSin θ_{OB} = 494.5 ft. and the ground roll distance S_g = 2500 – 494.5 =2005.5 ft. Using the (3.4) we determine (T/W) = 0.107 and relative velocity related to (T/W) is V_∞= 0.7(1.1V_{stall}) = 82.34 ft/ sec. Through this we can determine the power required during takeoff P_R = TV_∞=124578.7ft.lb/sec. This power required value helps to find out the shaft brake power, assuming the propeller efficiency as 0.85. Shaft break power, P = $\frac{P_R}{\eta_{pr}}$ =146563.26 ft.lb/sec or 266.48 hp.

To determine the aspect ratio of the airplane Coefficient of drag at zero lift is playing an important role. Wetted area to the wing reference area is approximately $\frac{S_{wet}}{S_{ref}}$ = 4 as per historical data of a single engine general aviation aircraft. From the historical data of different aircraft the coefficient of skin fraction C_{fe} is 0.0042 and the Reynolds number related to it is approximately 10^7. Now the Coefficient of drag at zero lift ${C_D}_0 = \frac{S_{wet}}{S_{ref}}\times C_{fe}$ = 0.017. Drag polar equation is $C_D = {C_D}_0 + kC_L^2$. The coefficient of k = $\frac{1}{4C_{d_0}((L/D)_{max})^2}$ = 0.075. For a low speed aircraft Oswald efficiency is 0.6. Aspect ratio, AR = $\frac{1}{\pi e_0 k}$ = 7.07.

VI. PERFORMANCE PARAMETERS OF THE AIRPLANE

In this study of conceptual design process performance parameter of a low speed propeller driven aircraft is estimated and mentioned in the table below:

Maximum coefficient of Lift ($\mathbf{C_{Lmax}}$)	2.115
Wing loading ($\frac{W}{S}$)	26.5
Thrust to weight ratio ($\frac{T}{W}$)	0.107
Lift to Drag ratio ($\frac{L}{D}$)	14
Gross weight during takeoff(W_0)	14140 lb
Fuel Weight (W_{Fuel})	2194 lb
Fuel tank capacity	390 gal
Wing area (S)	533.6 ft^2
High lift device	Single slotted trailing edge flap
Drag coefficient at Zero Lift(C_{D_0})	0.017
Aspect Ratio (AR)	7.07
Engine power required during takeoff (P_R)	124578.7 ft.lb/sec
Shaft break power (P)	266.48 hp

Table 1: Performance parameter of a low speed propeller driven aircraft

VII. CONCLUSION

From this conceptual design process of the aircraft we estimated the gross weight of the aircraft and the fuel weight and fuel tank capacity of the aircraft to cover required range. Gross weight of the airplane plays a vital role to determine the wing loading and thrust to weight ratio.

In this study from the numerical analysis of the NACA 23012 and NACA 23015 it can be observed that using a geometrically twisted wing produce more lift and structural strength than using a single airfoil shape configuration. From the fig. 5 and fig. 6 it can be observed that NACA 23012 produce more lift than NACA 23015 at same angle of attack. From the calculation of $\frac{W}{S}$ it is clear that if $\frac{W}{S} < 37.21$ lb/ft^2 then the landing distance is less than 2200ft which is satisfying the requirements of the airplane. It can be observed that the initial data before manufacture an airplane is estimated in this conceptual design process of a low speed propeller driven aircraft.

VIII. REFERENCES

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