CONCEPTUAL DESIGN OF HIGH PERFORMANCE UNMANNED AERIAL VEHICLE

Varsha N[1], Somashekar V[2]

[1]Assistant Professor, Department of Aeronautical Engineering, Acharya Institute of Technology, Bangalore – 560107, India

[2]Assistant Professor, Department of Aeronautical Engineering, Acharya Institute of Technology, Bangalore – 560107, India

E-mail: varshan@acharya.ac.in

Abstract: In current day's Unmanned Aerial Vehicle (UAV) are widely used in every field, almost from military till the commercial purpose. Usage of UAV has decreased the burden on the human, where the manpower and risks during critical conditions (war fields) are reduced. Therefore the demand for the development of the unmanned aerial Vehicle is high. Interpreting the conceptual design data of UAV is difficult because of lack of availability of their data sheets. This paper is addressed to develop a conceptual design process of high-performance UAV, which carries a maximum payload of about 300kg and can travel for the maximum range 900km/hrs. For about 50 hours of maximum endurance. Where these three parameters are considered as the main requirements. By drawing constraint diagram, feasible design space for the aircraft was frozen; using which initial sizing of the wing, wing airfoil selection, and a suitable power-plant selection was done. Fuselage design was carried out looking at the available literature on the existing aircrafts. Propeller design was done to match the thrust requirements obtained from the constraint diagram. Empennage design was done to achieve the desired static margin of the aircraft. This process completes the conceptual design of UAV, where designed aircraft meets the requirement. The outcome of this paper enhances the understanding of the conceptual design process for the academicians as well as the researchers.

Keywords: High Maneuverability, Constraint Diagram, Payload, Aerodynamic Efficiency, Conceptual Design, Wing loading (W/S), Thrust loading (T/W), Maximum take-off weight.

Nomenclature

L Lift
D Drag
T Thrust
W Weight
C_l Co-efficient of lift
C_d Co-efficient of drag
u Mean flow velocity, m/s
v Velocity, m/s
ρ Density, kg/m^3
μ Viscosity of air, kg/m-s
AR Aspect ratio
S_g Take-off length, m

Abbreviations

UAV Unmanned Aerial Vehicle
CFD Computational Fluid Dynamics
Re Reynolds number
MTOW Maximum take-off weight

1. INTRODUCTION

Nature has been the design driver for majority of man-made inventions; one such is the aircraft. Looking at the birds fly in the sky, desire of flight in man was invoked. From the days of first flight of Kitty Hawk by Wright Brothers in 1906 [1] up to the current day, design and development of aviation industry is improved tremendously. Now a day's unmanned aerial vehicle (UAV) is the fascinating concept in Aeronautical field which is used in almost all the fields.

The ultra-light UAV are meant to perform the complex makeovers in combat field as it gives the better performance compared to the others. Dueto this fact, usage of UAV is getting more important in both the combat and commercials fields, with the ability to do task at a feasible cost. Designing the high performance UAV is a challenging process as it involves meeting up all the design constraints or customer requirements by keeping the lowest possible take-off mass. Also that the maximum take-off mass of UAV category aircrafts should not be more, which invokes for use of advanced light weight engineering materials that makes design process more interesting and challenging. The manoeuvring capabilities of the current designed UAV is to achieve +5g to -4 g turns. Hence the current plane will be a trend setter design with such high manoeuvring capabilities.

This paper deals with providing a concept design of unmanned aerial vehicle with the fixed wing.

2. METHODOLOGY

The requirements for UAV were stated by thoroughly understanding the literatures that are reviewed before starting to design.Few methods are carried out to design the UAV, which starts from plotting the constraint diagram based on the requirements posed in order to select the design space. Initial sizing of the plane parts like fuselage, empennage was carried out based on thumb rules and empirical formulae available in the literature. Based on W/S value in constraint diagram, wing airfoil selection is made in order to achieve maximum C_l and wing designing is carried out. Propeller selection is made based on the T/W value obtained in design space of constraint diagram and wing airfoil analysis was carried out using low fidelity inviscid codes to ensure the wing designed from this airfoil produces the desired lift throughout the mission profile.

3. DESIGN PROCESS

Design specification/ Requirement – In order to design an aircraft there should be some requirements where the designed UAV meets all specifications. During the design process few parameters are assumed, and few are chosen from the literature survey that is made. Currently for this design, RQ-1 Predator aircraft is referred and few specifications are chosen this aircraft design [2].

Design Requirements are -

Takeoff distance and landing distance = 100m to 200m
Cruise distance = 37.5 m
Endurance = 8-10 hours
Maximum Takeoff weight = 500Kg
Cruise velocity = 55m/s
Range = 1000m

Generation of constraint diagram

Constraint diagram analysis was carried out by equating each of the design constraints as a function of thrust loading (T/W) and wing loading (W/S). Then constraint boundaries were set up on a graph as thrust to weight ratio (T/W) against wing loading (W/S). The space bounded by the constraint boundaries was identified as the design space. After four iterations, the most feasible combination of

T/W and W/S values were chosen for power-plant selection and wing design respectively for the aircraft [3].

Take-off performance

The take-off runway length (Sg) value is considered is 500m for this, W/S values corresponding T/W values were calculated using the formula as shown below [4].

$$Sg= \frac{1.21 W/S}{g\rho_{\infty} C_{Lmax} T/W}$$

To be on the safer side, take-off distance was chosen to be 250 m, was considered to estimate the T/W values. Since the information regarding the wing airfoil and the type of high lift device(s) used in the plane is not available at this point of design stage, the maximum lift coefficient value of the plane during take-off (Cl max) was assumed to be 1.6 [4].

Landing performance

For the given landing runway distance, W/S value was calculated using the formula as shown below[4].

$$\frac{W}{S} = \frac{9.81 * (S_g - 121.6) * (C_{l\max}) * (\rho/\rho_{sl})}{7.293}$$

To be on the safer side, for the design, landing distancewas chosen to be 100 m, which is 1% lower than the actual design constraint specified (200 m) was considered to estimate the W/S value. Since the information regarding the wing airfoil and the type of high lift device(s) used in the plane is not available at this point of the design stage, the maximum lift coefficient value of the plane during landing (C_{Lmax}) was assumed to be 1.6.

Cruise and ceiling performance

Since there was no design constraint specified on the cruise Mach number, the plane was assumed to cruise at an altitude of 7000m above sea level at 55 m/s. Service ceiling of the plane was assumed to be 5 km above sea level. This choice was made on the cruise altitude. Based on the manipulation of energy state (kinetic and potential) and the available excess power (Ps), the cruise curve of the plane was computed [5]

$$PS = \frac{(T-D)V}{W}$$

$$\frac{TMSL}{W} = \frac{\frac{\beta}{\alpha} * \frac{q}{\beta} * \frac{Cd0}{W/S} + \frac{\beta}{\alpha} * K * n^2 * \frac{W}{S}}{\frac{q}{\beta} + \frac{1}{V} * \frac{dh}{dt} + \frac{1}{g} * \frac{dv}{dt}}$$

Where;

Thrust lapse rate = α = T/T MSL

Aircraft mass fraction = β = W/WTO,

TMSL is the static sea-level total engine thrust in N

Take-off mass in N

n = load factor during cruise = 1

For the present aircraft;

IOP Conf. Series: Materials Science and Engineering **376** (2018) 012056 doi:10.1088/1757-899X/376/1/012056

α = T/TMSL = (ρ2.5km/ ρSL) 0.8 = (0.957/ 1.225)0.8 = 0.821

Based on the knowledge of similar class of existing UAV, wing aspect ratio for the plane was assumed to be 8[2] and the mass fraction of the pane at cruise was assumed to be 0.993. Since the CFD results were not available at this point of design, zero lift drag coefficient (CD0) of the plane was assumed to be 0.03. To ensure that the lift distribution over the wing is close to that on an elliptic wing (ideal), span efficiency factor (e) of the wing was assumed to be 0.9.

Now considering constant altitude, constant speed cruise, and T/W value for various W/S values were calculated.

Carpet plots – design trade-off

The number of design variables to be considered during the design of the aircraft is at least 10 and sometimes more than 50 [3]. As a designer it is a difficult task to sort out through all possible combinations in a systematic fashion to find the most feasible combination. In this regard, trade off studies was carried out as a part of design, in order to know the range of feasible T/W and W/S combinations that has to be incorporated in the current design. Several design parameters like the wing aspect ratio, flying speed, flying altitude, wing area, wing loading values were considered, multiple combination of these values were presented on trade study charts called the Carpet plots.

Therefore from the constraint diagram figure-4,the design space chosen is with T/W = 0.23 and W/S = 300.

Design of wing

W/S value corresponding to the design point chosen based on the constraint diagram section used to perform initial sizing of the wing.

Wing loading (W/S) obtained from the constraint diagram was 300 N/m2. Knowing the maximum take-off mass of the plane is 500 Kg we will consider as 550 kg just to make sure that designed plane would be able to take up the increased in MTOW, wing area (S) was calculated as:

$$\frac{W}{S}\text{=300=> S = 17.985 m2}$$

The aspect ratio (AR) of the wing of the plane was assumed to be 8 after the analysis of carpet plots. Using the aspect ratio and wing area values, wing span (b) was calculated which is 12.The plane is expected to cruise at 7 km altitude above mean sea-level with a cruise speed of 55 m/s .Taper ratio (λ) for an UAV ranges between 0.2-0.3 [2]. For the current design, Taper ratiois 0.3. Now, the root chord (Cr), tip chord (Ct) and the mean aerodynamic chord (mac)of the wing were estimated as

Root chord Cr = 2b/(AR(1+λ)) = 2.44 m

Tip chord = λ = Ct/Cr => Ct = 0.498 m

Mean aerodynamic centre mac= 2/3 Ct(1+ λ + λ 2)/(1+ λ) = 1.721 m

Mid-wing configuration was chosen. Since the current design is on high performance plane. The wing spar can be cut in the half in order to save space inside the fuselage. The aerodynamic advantages of choosing a mid-wing configuration is it's aerodynamically streamlined compared to the other configurations and it has less interference drag than the low wing or high wing configurations [6].

A dihedral angle for the wing was considered keeping in mind the lateral stability of the aircraft. The lateral stability is mainly the tendency of an aircraft to return to its original trim level-wing flight condition if disturbed by a gust and rolls around the longitudinal axis of the plane. By looking at the surveyed data of the existing aircrafts, a dihedral angle of $2^0$was considered.

Cruise mass of the plane was estimated considering the take-off mass and cruise mass fraction [7].

Wcruise = WTO* β * 9.81 = 5034 N

At cruise condition, the design lift coefficient (Cl) of the plane was calculated as:

(Cl) plane = Wcruise /(Area of wing * q)=> Cl = 0.6745

The design lift coefficient that the wing has to achieve at cruise was approximated as:

(Cl) wing = Clplane of the plane at cruise/0.95= 0.71006

Further the design lift coefficient that the wing airfoil has to achieve at cruise was approximated as:

(Cl) airfoil = Cl of the wing at cruise/0.9= 0.7889

Details on the use of high lift devices are not decided at this point and hence the maximum lift coefficient (Cl) that the airfoil should yield is unknown. So, NACA airfoil sections that yield an ideal lift coefficient of at least 0.8 with maximum 'CL max' value were looked for. Though the calculated ideal lift coefficient value was 0.78, once the entire wing is built, the effective ideal lift coefficient value reduces due to 3D effects. Hence an airfoil with a higher ideal lift coefficient value than the estimated value was chosen. NACA 631-308 that yields an ideal lift coefficient value of 0.854 at 8^0wing setting angle was chosen. The reasons for the choice of this airfoil are because it has the lowest Cd min value of 0.00592, with a (Cl / CD) max value of 140 and moderate stall quality.

The surface model of the top view of the full wing generated in Catia V5 environment is represented in Figure 1.

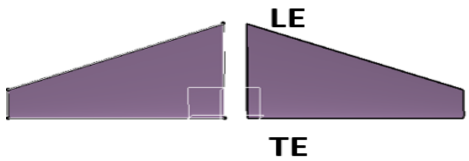

Figure 1: Top view of the wing

Selection of power plant

The most important criteria to select the engine type relates to the aircraft performance. The plane is designed to cruise at 2.5 km altitude above the sea-level with a cruise speed of 55 m/s (0.1664 Mach). As a designer, putting a constraint on the absolute ceiling of the plane is 5 km above mean sea-level, turbo-prop and piston prop engines are the most suitable means of propulsion.[8] Although, there exists a weight penalty in using turbo-prop due to the additional reduction gear unit. Furthermore, piston engines have the best propulsion efficiency at lower altitudes than turboprop engines [9].

From the constraint diagram, thrust to weight ratio (T/W) of corresponding to the most feasible design point was identified to be 0.23. The total take-off thrust the engine has to generate was calculated as [10]

Tstatic = T/W (from Constraint diagram)*Weight of Aircraft

Tstatic = 2.76 kN

Power required from the engine to power the flight was calculated as:

Prequired = Tstatic * Vmax

Prequired = 2.76 × 61.72 = 170.34 kW

Assuming a ηtransmission of 90% and ηpropeller of 80%, the actual power required from the engine was calculated as:

Actual Prequired = Prequired / (η transmission * ηpropeller) = 170.34 / (0.9 × 0.8) = 236.58 kW

IOP Conf. Series: Materials Science and Engineering **376** (2018) 012056 doi:10.1088/1757-899X/376/1/012056

Design of propeller

The constraint diagram shows the thrust required for the aircraft to operate at given conditions. A suitable propeller has to be selected from the available manufacturers which can deliver the thrust required. The details of the propeller geometry were obtained from manufacturers such as Hartzell®, Airmaster® etc. The propeller geometry specifications such as blade angles, thrust generated were not provided by the manufacturers. Therefore it was decided to design a propeller.

The propeller was to be designed based on the certain constraints. The RPM of the propeller was obtained from the engine manufacturer at full power. Another constraint was the tip Mach number which was not to exceed 0.8 as there would be a huge shock losses if the flow reaches transonic stage [11]. The power available from the engine was also fixed. The Input data for design of propeller is listed below

Input data for design of propeller

Rotational Speed=2000-2600(max)
Power available from engine=260kW
Relative Tip Mach no. not to exceed 0.8
Blade Loading=16-24 kW/m2

The design of propeller usually starts with selection of diameter of the propeller. After initial survey of existing propellers in UAV, it was decided that three bladed propeller of 2.4 m diameter would be optimum. The blade loading factor which gives an estimate of the power absorbed by the propeller per area of the blade is to be around 16 to 24 kW/m2.

Blade loading*n = (Power available from engine)/(Propeller Disc area) , Where n = no. of blades

But it was found that for the propeller diameter of 2.4 m and three blades, the blade loading factor was 24. In order to be on a safer side the blade loading had to be reduced. To reduce the blade loading either the power absorbed from the engine by the propeller has to be reduced or the propeller disc area has to be increased. Since the propeller and the engine output shaft are directly coupled by a common shaft. Reducing the power (lowering the RPM) hinders the thrust required for the airplane. Hence increasing the area of the propeller is an option, but doing so will increase the diameter. Increase in diameter will in turn affect the tip Mach no which is not to exceed 0.8 [12]. Therefore to decrease the diameter by increasing the chord length is preferred. Thus, a four bladed propeller was chosen with a diameter of 2 meters. Balancing is another fact for smooth operation of the propeller, three bladed propellers are statically imbalanced which tends to make the shaft bend on one side. Hence four bladed propellers barge in becoming the right choice for engineers.

From literature [2], for micro air vehicle Eppler 193 and combination of NACA series near the hub have been used. Since these operate at low Reynolds numbers, these aerofoils cannot be used. Therefore with further investigation from [6] studies carried out for higher diameter propellers, RAF 6 (Royal Air Force), Clark-Y aerofoils have been used from hub to tip. By considering these data, Clark-Y was selected in this design[13]

Design of fuselage

The primary design objective of fuselage of the current design is to accommodate the payload which may include the weapons or defence materials along with the sufficient amount fuel/ batteries for the requiredflight. By looking at the fuselage configurations of the existing aircrafts of the same class, the fuselage configuration considered for the current design is represented

Sizing of fuselage

The current design is a UAV aircraft. If circular cross section is incorporated, at the tail end of the fuselage, the structure will become weak for empennage attachment. As a design choice, the fuselage cross section was selected to be non-circular in shape as per NASA report [14].

IOP Conf. Series: Materials Science and Engineering **376** (2018) 012056 doi:10.1088/1757-899X/376/1/012056

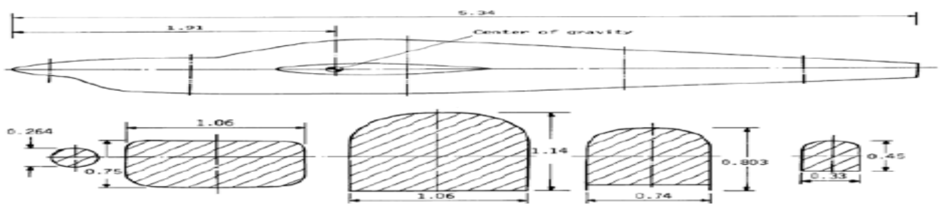

Figure 2- Fuselage geometry of NASA plane [14].

A rectangular cross section with round corners was employed in order to avoid sharp corners in the geometry and hence reduce the probability of flow separation at moderate angles of attack or sideslip. Fuselage geometry was determined based on the position of cockpit and length of moment arm required to generate tail moment. Overall fuselage length (Lf) depends on two parameters; namely the fuselage width (W) and fuselage height (H). The rear portion provides attachment of tail surfaces. The slenderness ratio for the present design was estimated as:

Slenderness ratio = Lf/Df =7200/1300=5.54

Design of horizontal tail

Horizontal tail is the stabilising surface that stabilises the aircraft along the longitudinal axis of the plane. The range of values of the non-dimensional cg limit (Δh), which is the difference between the most forward and the most aft position of the aircraft cg is 0.1 to 0.3 for an aerobatic plane. Therefore, at cruise, a reasonable assumption for the value of h (Xcg/mac) at the early stage of the horizontal tail design would be about 0.2 [2] since the aircraft cg moves during cruising flight, the horizontal tail airfoil must be able to create sometimes a positive and sometimes a negative lift. This requirement necessitates the horizontal tail to behave similarly for both positive and negative angles of attack. For this reason, a symmetric airfoil section is a suitable candidate for a horizontal tail. Also knowing that the t/c ratio of the horizontal tail airfoil used in aerobatic planes should be between 8-10% from the survey of existing aircrafts of the same class NACA 0009 airfoil was chosen for the horizontal tail.

The wing-fuselage combination pitching moment coefficient at zero lift condition (Cm0_wf) was calculated as[15]

The horizontal tail volume coefficient (VH) and the tail efficiency factor (η_h) were assumed to be 0.6 and 1 respectively at this starting point of tail design [4]the value can be refined in the successive iterations of the tail design process. The optimum tail arm length (lopt) was estimated using the formula, Where Kc is the correction factor; it varies between 1 and 1.4 depending on the aircraft configuration. Kc = 1 is used when the aft portion of the fuselage has a conical shape. As the shape of the aft portion of the fuselage goes further away from a conical shape, the Kc factor is increased up to 1.4. For the current aircraft, a tadpole configuration fuselage was used. So, Kc value was assumed to be 1.1. Thus, lopt was calculated as:

$$l_{opt} = 1.1\sqrt{(4*17.985*12.72*0.6)/(\pi*1.3)} = 4.049 \text{ m}$$

The horizontal tail area (Sh) was calculated as:

$$VH=(Stl\_opt)/(S\_wc) \Rightarrow S= 3.756 \text{ m2}$$

The horizontal tail required lift coefficient (CLh) at cruise was calculated using the trim equation as:

$$Cm0\_wf + CLw\,(hcg - hac) - \eta_h\, VH\, CLh = 0 \quad = CLh = -0.0226$$

As a design choice, the tail aspect ratio was assumed to be two-third of wing aspect ratio [6]

$$ARh = 2/3 * ARw = 2/3 * 9 = 6$$

Using the horizontal tail aspect ratio and area values, tail span (bh) was calculated as:

bh = (ARh * Sh)0.5= (6 ×3.756)0.5 = 4.747 m

Horizontal tail taper ratio (λh) for an aerobatic plane ranges between 0.4-0.7 [6]. For the current design, a value of 0.3 was used. Now, the root chord (Cr), tip chord (Ct) and the mean aerodynamic chord (mac or c ̅) of the tail were estimated as:

Cr = 2b/(AR(1+λ)) = (2*4.747)/(6(1+0.3)) = 1.217 m

λ = Ct/Cr=> Ct = 0.3*1.217 = 0.365 m

mac = 2/3 Ct ((1+ λ + λ2)/(1+ λ)) = 0.868 m

Design of vertical tail

The primary purpose of the vertical tail is to maintain the aircraft directional stability and directional trim. The current aircraft has symmetry about the xz plane, so the directional trim is naturally maintained. But there is always a slight asymmetry in the aircraft's xy plane.

The horizontal tail volume coefficient (Vv) and the tail efficiency factor (ƞv) were assumed to be 0.03 and 1 respectively at this starting point of tail design [4]. The value can be refined in the successive iterations of the tail design process. The vertical tail arm length (lvt) was estimated using the formula:

lvt = Kc √((4(bw S)_w (Vv) ̅)/(πD_f))

Where Kc is the correction factor; it varies between 1 and 1.4 depending on the aircraft configuration.

Kc = 1 is used when the aft portion of the fuselage has a conical shape. As the shape of the aft portion of the

Fuselage goes further away from a conical shape, the Kc factor is increased up to 1.4. For the current aircraft, a tadpole configuration fuselage was used. So, Kc value was assumed to be 1.4. Thus, lvt was calculated as:

lv = 1.4√((4*12.73*17.985*0.03)/(π*1.3))=3.631 m

The horizontal tail area (Sh) was calculated as:

Vv = (Svl*v)/(Sw*b_w)

V = (0.03*17.985*12.73)/3.631=1.891 sq.m2

The vertical tail aspect ratio was assumed to be 1 [6]

Using the vertical tail aspect ratio and area values, tail span (bv) was calculated as:

bv = (ARv* Sv)0.5= (1 × 1.891)0.5 = 1.375 m

Vertical tail taper ratio (λv) for aerobatic plane ranges between 0.4-0.7 [4]. For the current design, a value of 0.5 was used. Now, the root chord (Cr), tip chord (Ct) and the mean aerodynamic chord (mac or c ̅) of the vertical tail were estimated as:

Cr = 2b/(AR(1+λ)) = (2*1.375)/(1(1+0.5)) = 1.833 m

λ = Ct/Cr => Ct = ¬0.5 * 1.833 = 0.917 m

mac = 2/3 Ct ((1+ λ + λ 2)/(1+ λ)) = 2/3 *0.917*((1+ 0.5 + 〖(0.5〗^2)/(1+ 0.5)) = 1.426 m

As a design choice, vertical tail sweep angle at quarter chord was assumed to be 35^0.

The vertical location of the vertical tail relative to the fuselage centre line was calculated as:

H = lv * tan (αstall_w – i_w + 3) = 3.631 * tan (10 – 1.2 + 3) = 0.758 m

The design goal in high lift devices (HLD) is to maximize the capability of the wing.

Design of landing gear

Looking at the literature and as a design decision, tail dragger type landing gear system was incorporated for the present design [5. 16].

4. RESULTS AND DISCUSSION

Constraint diagram:

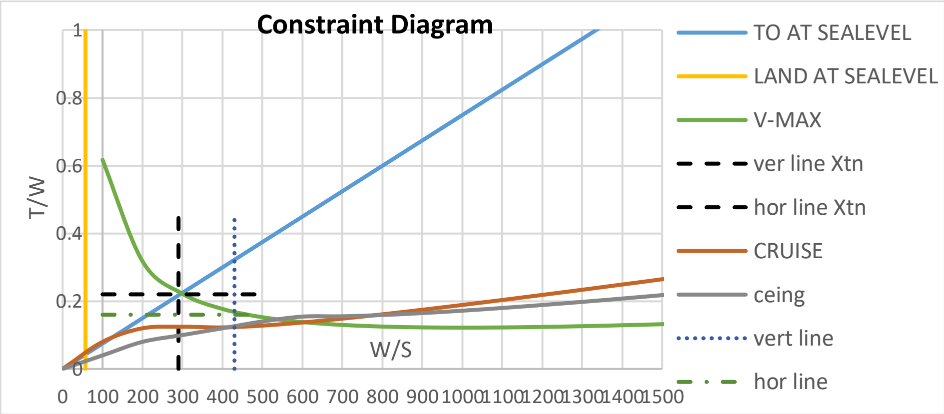

Figure 3 – Constraint diagram

The combination of constraints graphically identifies the feasible design space.In the current design, positive g-manoeuvre, maximum velocity and landing performance curves were observed to be the design drivers. Any combination of T/W and W/S values chosen within this region of design space will meet all the performance requirements. The design point chosen for the current design of UAV has been highlighted with a black lines. This particular point was chosen because it corresponds to the minimum value of T/W (to ensure the power plant size is kept as small as possible) and minimum value of W/S (to ensure manoeuvring capability of the wing). At the design point chosen.

T/W = 0.23 and W/S = 300 N/m2. This T/W and W/S values were used for power-plant selection and initial sizing of wing respectively.

Wing design

Based on the outputs of constraint diagram, wing design was done as an iterative process. NACA 631-308 was chosen as the wing airfoil. After initial sizing of wing, the geometry details of wing for last iteration are represented in table 1

Area of the wing	17.985	m2
Aspect Ratio	8	-
Wing span	12.723	m
Mean aerodynamic chord	1.55	m
Root chord	2.175	m
Tip chord	0.6524	m
Sweep angle at quarter chord	0	degree
Taper ratio	0.3	-

Taper angle	0	degree
Dihedral angle	2	degree
Wing setting angle	1.2	degree

Airfoil analysis

Initially, Panel code analysis was done using the Javafoil applet using 101 panels at the Reynolds number corresponding to cruise (527909) with Calcfoil as the stall model. The lift curve slope obtained is shown in figure 5. It was verified that the design lift coefficient of 0.8 was achieved at that Reynolds number corresponding to cruise phase.

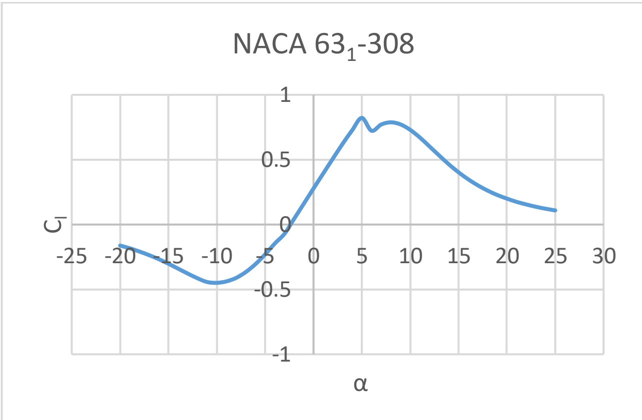

Figure 4 - Lift curve slope of NACA 631-308

The wing setting angle (iw) was initially determined to be the angle corresponding to the airfoil ideal lift coefficient. From figure 4, the airfoil ideal lift coefficient is 0.8, and the angle corresponding to the airfoil ideal lift coefficient was identified to be 1.2^0.

5. CONCLUSION

Typical design procedure for the unmanned aerial vehicle with high performance is calculated using the empirical relations and the data sheets are created based on the specifications. Conceptual design process goes on step-wise which gives the brief overview of how industries work in building and designing the aircrafts with the requirements posed at them by the customers. Each parts designing of the aircraft is done which help us to understand conceptual design concept. Designed aircraft is meeting the requirements to perform at higher altitudes of about 7000m for about 8-11 hours of endurance and to perform -5g and +6g turns which helps in combat flights.

References

[1]. L. Jenkinson J. Marchman 2003, "Aircraft Design Projects", Elsevier publishers
[2]. Air force Technology, http://www.airforce-technology.com/projects/predator-uav/
[3]. Daniel. P. Ramer 1992, "Aircraft Design – Conceptual Approach",AIAA,Inc.
[4]. J. D. Anderson 2011, "Aircraft Performance and Design", Elsevier publishers
[5]. Dr. Jan Roskam 1985, "Airplane Design", Roskam Aviation and Engineering Corporation
[6]. M.H Sadraey 2013, "Aircraft Design", Wiley publishers.
[7]. Reg Austin 2010, "Unmanned Aircraft Systems UAV design, development and deployment", Wiley publishers.
[8]. Leland M. Nicolai and Grant E. Carichner 2010, 'Fundamentals of Aircraft and Airship Design (AIAA Education Series)', Volume I, American Institute of Aeronautics and Astronautics.

[9]. Prof. Bhaskar Roy, Prof. A.M Pradeep,"Introduction to Aerspace propulsion", Available from: ¬< http://nptel.ac.in/courses/101101001/downloads/Intro-Propulsion-Lect- 29.pdf>.[5 January 2016]
[10]. S M Yahya 2006, "Turbines Compressor and Fans", Tata McGraw-Hill publishers
[11]. Durand. W. F, Lesley. E. P, "Experimental research on air propellers", NASA Technical reports Server, Available from: <http://ntrs.nasa.gov/search.jsp?R=19930091207>. [27 December 2015].
[12]. Prathapanayaka R, Vinod Kumar N, Krishnamurthy S J 2011, "Design, Analysis, Fabrication and Testing of Mini Propellers for MAV", Available from: <http://nal-ir.nal.res.in/10244/1/061_G08.pdf >. [3 January 2016]
[13]. http://www.drivecalc.de/PropCalc/, computer software 2016. Available from:< http://www.drivecalc.de/PropCalc/ > [January 2016].
[14]. Mohd. Ridhwan and B. Basuno 2014, "The Fuselage Model of Non Circular Section", Proceedings Of Data Mining, Civil and Mechanical engineering Conference. Available from: http://dx.doi.org/10.15242/IIE.E0214015. [20 December 2015]
[15]. Paul GerinFahlstrom and Thomas James Gleason, "Introduction to UAV system", Wiley Publications., 2012.
[16]. Nelson, R.C. "Flight Stability and Automatic Control", McGraw-Hill Book Co., 2007.