Design and Development of a Transitional Electric Unmanned Aerial Vehicle.

Ahmed Ashraf El-Sayed Gebril
Military Technical College, Egypt, engaash@gmail.com
Supervisor: Lt. Col. Dr: Ashraf Mohamed Kamal
Aircraft Mechanics Department, Egypt, ashraf.kamal@mtc.edu.eg

Abstract*– Transitional aircraft are systems that combine the advantages of fixed-wing airplanes (e.g., high speed, range and endurance) and rotorcraft (e.g., vertical take-off and landing, hovering) by flying in fixed-wing and rotorcraft flight modes and perform transition between these modes. Research in manned transitional aircraft started seventy years ago before being abandoned at the end of the last century due to many problems in design, performance, control, and stability. In the last decade, due to the advancement in the Unmanned Aerial Vehicle (UAV), transitional UAVs regained great interest in the aeronautical society. Motivated by their significant capabilities, and given a specific set of design requirements, this paper aims to present a complete methodology (including all necessary mathematical formulations) to develop a transitional UAV comprising the following crucial aspects (conceptual, preliminary, and detail design, aerodynamics, performance, and stability analysis). The required battery capacity and total mass are determined from mission analysis that include both VTOL and FW mission segments. The design is iteratively resized when the actual components of the propulsion system are selected.

I. Introduction

During the last decades, unmanned aerial vehicles (UAV) have been widely used in both military and consumer areas such as surveillance, tracking, monitoring and aerial photography. These UAVs are mainly classified into two types: fixed-wing UAVs and rotary-wing UAVs and each type has their own advantages and drawbacks. The general aim nowadays is to design an aerial system that integrates the advantage of both (i.e., VTOL UAV). like high speed, range and endurance, and of rotorcraft such as hovering, low-speed flight and Vertical Take-off and Landing (VTOL). Although UAVs have a great development, It’s still in its infancy and there is a huge space for the hybrid UAVs to become more mature.

A. Problem Definition:

The challenging problem is to design and manufacture a Transitional Unmanned Aerial Vehicle (VTOL UAV) with a constructed design requirement the UAV must obtain.

The VTOL UAV should satisfy these design requirements:

1) satisfy performance requirements:

such as Max Take Off Weight (5-30) kg, Payload Weight (3.5) kg, Range in FW Flight Mode (50) km, Endurance in FW Mode (3) hrs, Hover Endurance (5) min, Transition Time (≤ 8) sec.

2) Satisfy the following mission profile:

(VTO – transition from hovering to FW – FW climb – cruise – transition from FW to hovering – hovering – transition from hovering to FW – cruise back – transition from FW to hovering – Vertical landing).

3) Modular design

B. Platform design:

Transitional UAVs are categorized into two types: convertiplane and tail-sitter. A convertiplane maintains its airframe orientation in all flight mode, and different transition mechanisms are used to obtain cruise mode. On the other hand, a tail-sitter is an aircraft that takes-off and lands vertically on its tail, and entire airframe needs to tilt to accomplish cruise flight. There are a few subtypes for the main two types depends on the transition mechanism and airframe configuration.

C. Types of a Transitional UAVs:

Convertiplanes can be further categorized into four sub-types, including 1) Tilt-Rotors, 2) Tilt-Wings, 3) Rotor-Wings, and 4) Dual Systems. Tail-sitters can be classified into three types as follows: 1) Mono Thrust Transitioning (MTT),
2) Collective Thrust Transitioning (CTT)
3) Differential Thrust Transitioning(DTT). We note that the tilt rotor UAVs are more suitable and matches the design requirements than the other subtypes as in which the shafts and nacelles are only required to rotate rotors instead of wings or other heavy structures which saves power and weight. Moreover, due to their controllability and stability in vertical flight when compared to other hybrid UAVs.

So, we conclude that the tilt rotor UAVs are more suitable and matches the design requirements than the other subtypes as in which the shafts and nacelles are only required to rotate rotors instead of wings or other heavy structures which saves power and weight. Moreover, due to their controllability and stability in vertical flight when compared to other hybrid UAVs.

D. Flight dynamics modelling and control:

Developing a reliable flight dynamics model becomes more challenging compared with conventional aircraft. During the transition of tilt rotor or tilt wing the direction of thrust changes during the transition phase and therefore it should be modelled appropriately. The control system depends on the derived dynamics model that is highly complicated and nonlinear. The transition phase remains a critical part of the control system due to the multiple nonlinearities in the model. UAV models are nonlinear and linearized by applying relative equilibrium conditions around a steady state operating point. Although linear control loss or simple, easy, reduce efforts and

minimize design time but their performance degrades when operating away from the local equilibrium point.
The main reason of the transition mode complexity is that during changing from vertical flight to horizontal flight the operation is far away from the equilibrium condition. Current UAVs implement nonlinear controllers or three separate linear controllers for (vertical, horizontal and transition) modes. The most common linear control laws applied in hybrid UAVs is (PID and LQR) controller. The most common nonlinear control law is applied in hybrid UAVs is (gain-scheduling, back-stepping and NDI) controller.

II. Design procedures

Design process has mainly three phases conceptual design, preliminary design, and detailed design as shown in block diagram

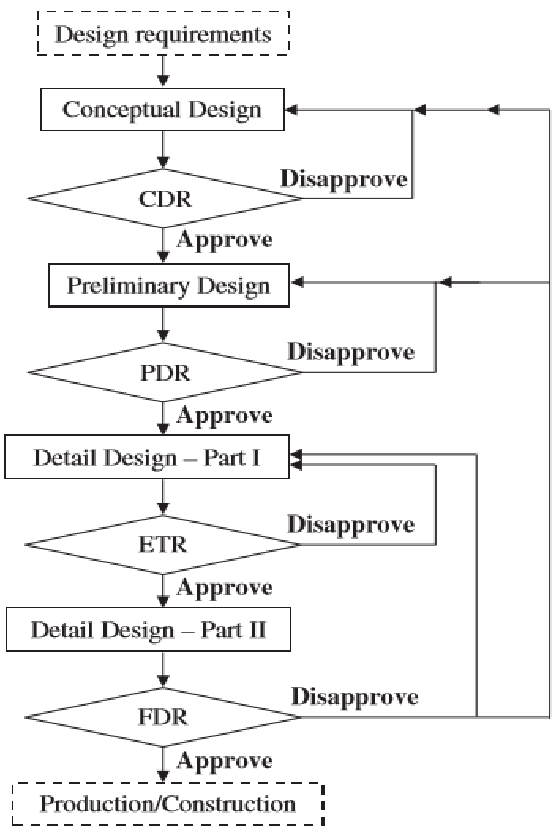
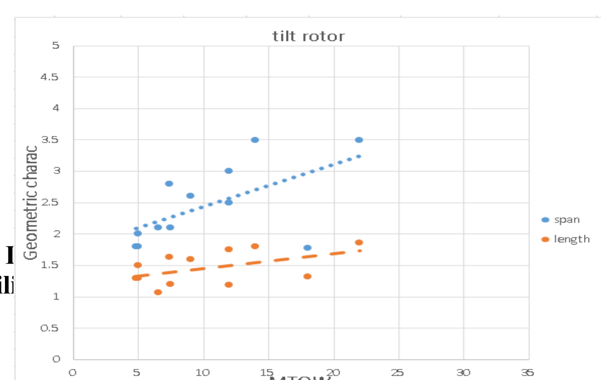


Fig. 1 show block diagram for design procedure.

A. Conceptual design:

1. Survey about transitional UAVs:

The most important step in the design process is doing a



survey on UAVs similar to the designed one in order to make a reference that tells us that we perform the calculations correctly. The output of this survey is a statistical data for different parameters that could be considered as a reference that show us the range about which the results are closes.
This statistical data presented in form of curve fitting.
These curves between different parameters of actual UAVs collected from survey and constructed on curve fitting show relations between these parameters and each other. As shown in

2. the transitional UAV configuration:

Each major aircraft component may have several alternatives which all satisfy the design requirements. However, each alternative will carry advantages and disadvantages by which the design requirements are satisfied at different levels. Since each design requirement has a unique weight, each alternative result in a different level of satisfaction, and a trade-off

Fig. 2 statistical data between geometric characteristics & TOW of tilt rotor.

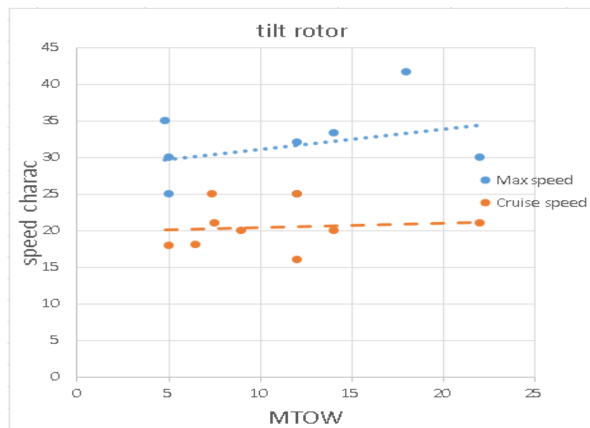


Fig. 3 statistical data between speed characteristics & MTOW of tilt rotor

analysis technique are used. This section reviews the configuration alternative for each major component. There is more than one configuration for wing, tail, fuselage, and propulsion system.
Then, the advantages and disadvantages of each component alternatives, plus the technique to select the best components configuration to meet design requirements, are presented in the following sections.
The primary impacts of each components alternatives are imposed on aerodynamic characteristics and performance such as (Stall speed, Maximum speed, cruise altitude, FW ROC, FW ceiling, Transition time, Helicopter ROC, Range, FW Endurance, Hover Endurance, etc.), fixed wing and hover stability, fixed wing and hover controllability, cost and ease of manufacture, payload capability, and modularity and transportation.
Each component alternative has a weight on the above consideration and there is a weight depending on design requirements and for each component the selected configuration is that has the largest weight when multiplying the ratio that represent the effect of each component

alternative on the prescribed parameters and the weight of these parameters that distributed according to design requirements. The output of wing configuration selection is a fixed wing with high position with tapered planform with a relatively small sweep back and zero dihedral, Empennage Configuration selection produces an aft conventional tail with one fin,
The selected fuselage is a single conventional with cross-section semicircular and semi rectangular. The output of propulsion system is a tractor electric motor propulsion system with tri-tiltable motor. The output of configuration selection is a free hand sketch show the primary shape of the main UAV components. As shown

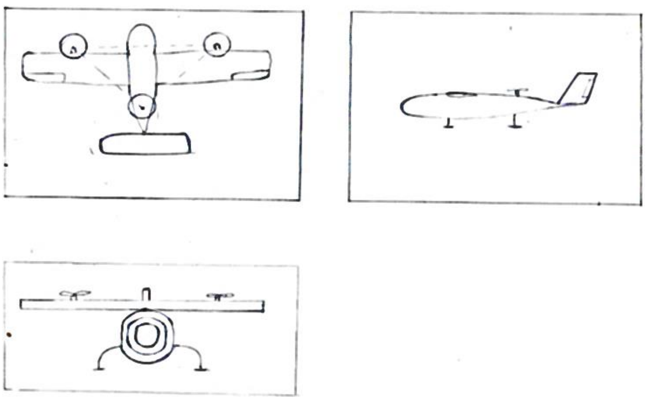


Fig. 1 show free hand sketch for initial configuration selection

3. Wing Area and Engine Sizing:

In addition to meeting range, endurance and cruise objectives. The UAV are usually designed to meet the performance objectives in both vertical flight mode and horizontal flight mode. The Sizing process steps are shown as follow

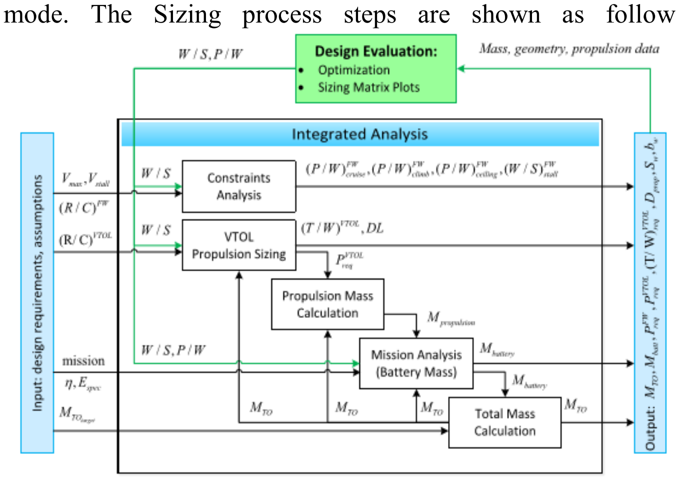


Fig. 2 show block diagram for sizing process.

This block scheme shows the FW-VTOL UAV sizing process. The classical performance equations of FW aircraft are used to allow obtaining a relationship function between wing loading(W/S) and the corresponding power loading(W/P). The four performance requirements used in sizing in FW mode are:
1) stall speed
2) maximum forward speed
3) maximum rate of climb (Roc)
4) maximum flight altitude (ceiling)
There are a few aircraft parameters (such as aircraft maximum lift coefficient, Aspect ratio, zero lift drag coefficient…etc.) which may be needed throughout the technique, but they have not been calculated analytically prior to this preliminary design phase. These parameters will currently be estimated based on the statistics.
1) stall speed:
One of the aircraft performance requirements is a limit to the minimum allowable speed. For most aircraft the mission demands a stall speed not higher than some minimum value, so It is important to consider such a limit in the design to prevent aircraft stall during its operation. The wing loading can be formulated as a function of the stall speed as given

$$(WL)_{st} = 0.5\rho_0 V_{st}^2 C_{L_{max}} \quad (1)$$

where: (WL) is the wing loading, ρ_0 is the air density at sea level, V_{st} is the required stall speed and CL_{max} is the aircraft maximum lift coefficient.
2) maximum forward speed:
The second considered performance requirement is the maximum forward speed. Both wing and power loadings contribute in achieving this parameter. the performance equations of propeller-driven FW aircraft are used to formulate a relationship that constrains the variation of the power loading as a function of the wing loading and the required maximum speed as given

$$(PL)_{v.max} \le \frac{\eta_p \eta_{tr} \sigma_{air}}{(0.5\rho V_{max}^3\, CD_O(WL)) + (2K(WL)/\, \rho V_{max})} \quad (2)$$

where: PL is the power loading, $\eta_p$is the propeller efficiency, CDo is the aircraft zero lift drag coefficient, σ air is the relative air density/de-rating factor, ρ is the air density at the altitude of maximum speed, V_{max} is the required maximum speed, AR is the wing aspect ratio, e is the Oswald efficiency, and
K = 1⁄π.e. AR.
3) maximum rate of climb (RoC):
Another performance requirement is to climb in fixed-wing flight mode by a certain ROC. Similarly, the climb performance equations of propeller-driven aircraft are formulated to give the relationship between the power loading as a function of the wing loading and ROC as given

$$PL)_{ROC} \le \frac{\eta_{tr}}{(V_y/\eta_p) + (1.155/(L/D)_{max}\eta_p)\sqrt{2(WL)/\, \rho_0\sqrt{3\, CD_O/K}}} \quad (3)$$

where: (L/D) max is the maximum lift to drag ratio and Vy is required rate of climb.

4) maximum flight altitude (ceiling):
Ceiling in fixed-wing aircraft, the highest altitude that an aircraft can safely fly straight and level, is another requirement that affect wing and power loadings. In this formulation, we deal with the service ceiling instead of absolute ceiling to allow straight and level flight with a good stability margin at ceiling altitude. A typical value for the ROC at service ceiling

for low subsonic aircraft is $V_{y.c}$=0.5 m/s. Since the ceiling requirements are defined based on the ROC requirements, maximum rate of climb equation is reformulated at service ceiling parameters as shown

$$\mathrm{PL})_c \leq \frac{\eta_{tr}\sigma_{air}}{(V_{y.c}/\eta_p) + (1.155/(L/D)_{max}\eta_p)\sqrt{2(\mathrm{WL})/\rho_c\sqrt{3\,CD_O/K}}} \quad (4)$$

where: $V_{y.c}$is the rate of climb at service ceiling and ρ_c is the air density at service ceiling.

Sketch all derived equations in one plot. The horizontal axis is power loading (W/P) and the vertical axis is wing loading (W/S). Thus, the plot illustrates the variations of power loading with respect to wing loading. These graphs will intersect each other at several points and may produce several regions. Identify the acceptable region inside the regions that are produced by the axes and the graphs. The acceptable region is the region that meets all aircraft performance requirements. Determine the design point (i.e., the optimum selection). The design point on the plot is only one point that yields the smallest engine in terms of power (i.e., the lowest cost). From the design point, obtain two numbers: corresponding wing loading (W/S)d and corresponding power loading (W/P)d. Calculate the wing area and engine power from these two values, the wing area is calculated by dividing the aircraft take-off weight that determined in the next design step by the wing loading. The engine power is also calculated by dividing the aircraft take-off weight by the power loading:

$$s = W_{TO}/(\frac{W}{S})_d \quad (5)$$

$$p = W_{TO}/(\frac{W}{p})_d \quad (6)$$

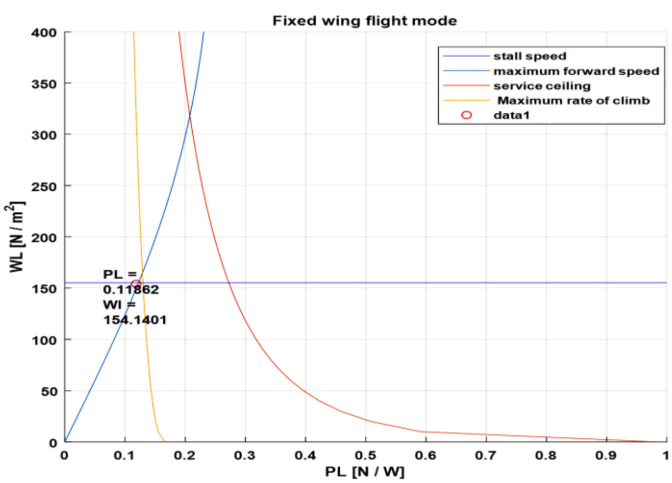


Fig. 3 Fixed wing matching curve.

2. In vertical flight mode:

The traditional rotorcraft performance equations of helicopter aircraft are used to allow obtaining a relationship function between power loading(PL) and the corresponding disc loading(DL). The five performance requirements used in sizing in vertical flight mode are:

1) Hovering Flight.
2) Vertical Climb Flight.
3) Hover Ceiling.
4) Wingspan Limit
5) Transition Flight Sizing.

there is a larger number of performance requirements that contribute to engine power and rotor size (e.g., the requirements for hovering at certain altitude, executing a vertical climb speed, hover at specific ceiling altitude, performing forward climb flight, performing some maneuvers, and descent flight).

1) Hovering Flight:

During rotorcraft operation, a large number of transitional UAVs have the rotors over the wings that needed for fixed-wing operation. his aspect complicates the design process. the preliminary design stages, analytical equations can be used to estimate this download force. The required increase in the rotor's thrust to overcome the download force/vertical drag can be expressed as

$$\Delta T = [f_w(S/A)]T \quad (7)$$

$$f_w = k_{wet}\left(\frac{n^2 CD_{wet}}{4}\right) \quad (8)$$

where T is the rotor thrust, S is the wing area, n is a factor depending on the vertical distance between the wing/fuselage and the rotor (has a value from 1 to 2), CD_{wet} is the drag coefficient based on the area wetted by the rotor's downwash, and k_{wet} is the expected ratio of the wing/fuselage area wetted by the rotor's downwash to the total wing area in the rotorcraft flight mode. The parameter k_{wet} depends on the selected aircraft configuration and can be estimated based on data for similar aircraft. For simplicity, the term "n^2CD.wet / 4" can be approximated by a value of 0.7.

The power loading constraint, in hovering, as a function of disc and wing loadings is obtained from

$$\mathrm{PL})_{hov} \leq (\mathrm{FM})\,\eta_{tr}(1\text{-}f_w[(\mathrm{DL})/(\mathrm{WL})])^{\frac{3}{2}}\sqrt{2\,\rho/(\mathrm{DL})} \quad (9)$$

where,

FM is the Fig of merit (FM), which is the ratio of ideal induced power for a rotor in hover obtained from the momentum theory and the actual power consumed by the rotor.

$$\mathrm{FM}=\frac{P_i}{P} \quad (10)$$

For simplicity it assumed to equal 0.7.

2) Vertical Climb Flight:

The second aspect in the proposed rotorcraft design aimed toward HA is to consider the required vertical climb flight. As there is no asymmetric flow in purely vertical climb on the rotor blades, the same expression for the download force/vertical drag in hovering can be used in vertical climb

$$\Delta T_1 = f_p S V_y^2 \quad (11)$$

$$f_p = 0.5\rho_0 k_p C_{D.p} \quad (12)$$

Where k_p is the expected ratio of the total aircraft projected area in the horizontal plane to the total wing area. The value of k_p depends on the selected aircraft configuration and can be estimated based on data for similar aircraft (for tail-sitters, $k_p$is approximated by 1, whereas initial estimate for other HA may vary from 1.2 to 1.5). $C_{D.p}$ is the drag coefficient based on the aircraft projected area in the horizontal plane. For tail-sitters, $C_{D.p}$is approximated as the aircraft drag coefficient during cruise, whereas for TRA, $C_{D.p}$can be approximated as the drag coefficient for a similar size flat plate area (with a value between 1.17 for low aspect ratios to 1.45 for high aspect ratios)

The relation between the required power and disc loadings during rotorcraft vertical climb is obtained as follow

$$PL)_{V.C} \leq \frac{\eta_{tr}}{([f_p V_y^2 + (WL)]/[(WL) - f_w(DL)](2-k_i)(V_y/2) + k_i\sqrt{(V_y/2)^2 + ((DL)/2\rho_0)([f_p V_y^2 + (WL)]/[(WL) - f_w(DL)])} + (\rho_0 V_{tip}^3/(DL))(\sigma C_d/8)}$$

Where, C_d is the average blade drag coefficient, σ is the main rotor solidity ratio (ratio between total blades area and rotor disc area), and V_{tip} is the rotor's tip speed, k_i is the induced power correction factor that accounts for any rotor tip losses, non-uniform inflow, wake swirl, non-Ideal wake contraction, and the finite number of blades. It has a typical average value in hovering of 1.15.

3) Hover Ceiling:

the service ceiling will be used in the performance equations instead of the absolute ceiling to allow hovering flight with a stability margin at the desired ceiling altitude. A typical value for the ROC at service ceiling for low subsonic aircraft is V_{yc}=0.5 m/typically, for tiltrotor UAVs.

Typical electric motors are designed to operate at sea-level ambient temperatures and below 1000m. For altitudes above 1000m, the output power is affected by the motor's ability to dissipate heat due to the reduction in the surrounding air density. To account for such reduction in motor's power, a factor called "de-rating factor", usually specified by the motor's manufacturer, is used at some specific altitude ranges above 1000m. the symbol of σ air is also used to denote the de-rating factor if electric motor is used.

the relation between the power and disc loadings at the hover service ceiling altitude can be expressed as

$$PL)_{h.C} \leq \frac{\eta_{tr}}{(WL)/[(WL) - f_w(DL)][(2-k_i)(V_{yc}/2) + k_i\sqrt{(V_{yc}/2)^2 + ((DL)/2\rho_c)((WL)/[(WL) - f_w(DL)])}] + (\rho_c V_{tip}^3/(DL))(\sigma C_d/8)}$$

where: ρ_c is the air density at the service ceiling altitude.

4) Wingspan Limit:

This constraint is included only for configurations that employ more than one non intermeshing/non coaxial rotor along the wingspan. This constraint ensures that the rotors can be fixed

within the available wing span limit and do not collide with each other or with the fuselage during their operation in all flight modes. Based on the selected value of the wing loading in the FW mode, the wing's aspect ratio, the number of rotors, and the fuselage width at the wing root, the minimum allowable value of the disc loading that satisfy the wingspan limit is expressed as

$$(DL) \geq \frac{k_c^2(2n-2)^2(WL)}{n\pi(AR)\,k_{fuselage}^2} \qquad (13)$$

$$k_{fuselage} = 1 - \bar{w}_{fuselage} \qquad (14)$$

Where n is the number of rotors(n>1), kc is a factor to account for the required clearance between the adjacent rotors' tips (as a function of the rotor radius),

and W fuselage is the ratio between is the fuselage width at wing root and the wingspan. The values of kc and W fuselage can be obtained from historical data for similar aircraft (initial estimate may vary from 1.1 to 1.3 for kc and between 0.25 and 0.35 for W fuselage).

5) Transition Flight Sizing:

Transition sizing constraint is important for aircraft that require performing transition from VTOL to FW at specified time without losing altitude, and/or perform the transition at specific rotor's tilt angle. This is important to ensure that the aircraft has the adequate power to perform the transition at the specific requirements. To formulate the equations necessary for aircraft preliminary sizing during the transition flight mode, it is required to account for the following power terms: induced, blades' profile, parasite, and climb. The climb term is only needed if there is a change in the aircraft's altitude during transition.

Induced power calculated from:

$$P_{i.Nonideal} = k_i(Tv_i) = \frac{k_i T^2}{2\rho A V_{wb}} \qquad (15)$$

k_i has an approximate value in forward flight of 1.2 (somewhat higher than that in hovering), and V_{wb} typically has a value between 1.1 and 1.3 of the stall speed at the transition altitude.

As it is difficult to develop an accurate analytical model for the rotor's wake during transition to simplify the preliminary sizing the rotor's download force over the aircraft wing/fuselage is neglected.

the small vertical component of rotor profile drag is ignored.

the power required to overcome rotor profile drag is given as

$$P_o = \rho A V_{tip}^3 \left(\frac{\sigma C_d}{8}(1+4.6\mu^2)\right) \qquad (16)$$

on the plot is only one point that yields the smallest engine in terms of power (i.e., the lowest cost). From the design point, obtain two numbers: corresponding power loading (W/P) vt and corresponding Disc loading (W/P) vt

$$PL)_{tr} \le \frac{\eta_{tr}\sigma_{air}}{[k_i/\sin(\theta_{tr}))\sqrt{(-V_{wb}^2/2) + \sqrt{(V_{wb}^2/2)^2 + ((DL)/2\rho\sin(\theta_{tr}))^2}} + [(\rho V_{tip}^3/(DL))((\sigma C_d/8)(1+4.6\mu^2))] + ((0.5\,\rho V_{wb}^3 C_{Do}/(WL)) + (2K(WL)/\rho V_{wb})) + (V_{wb}^2/2gt_{tr})}$$

where μ can be expressed as:

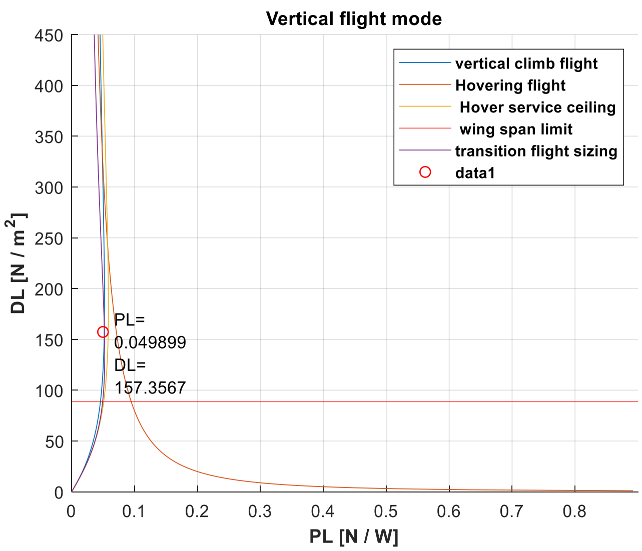


Fig. 4 Matching curve in vertical flight mode.

$$\mu = \frac{V_{wb}\sin(\theta_{tr})}{V_{Tip}} \qquad (17)$$

including power required to overcome the wing's induced drag corresponding to wing-borne velocity, is represented by:

$$P_p = DV_{wb} = 0.5\rho V_{wb}^3 S C_{Do} + \frac{2KS(WL)^2}{\rho V_{wb}} \qquad (18)$$

the required power to accelerate the aircraft from zero to wing-borne speed is considered and is given by

$$P_a = \frac{WV_{wb}^2}{2gt_{tr}} \qquad (19)$$

the constraint relation to obtain the necessary power loading for transition as a function of disc loading, wing loading, required transition tilt angle, and transition time can be deduced as given in above eqn.

Sketch all derived equations in one plot . The horizontal axis is power loading (PL) and the vertical axis is wing loading (DL) at different values of wing loading and rotor tilt angles. Using the wing loading obtained from the FW performance formulations. These graphs will intersect each other at several points and may produce several regions. Identify the acceptable region inside the regions that are produced by the axes and the graphs. The acceptable region is the region that meets all aircraft performance requirements. Determine the design point (i.e., the optimum selection). The design point

Calculate the rotor size and motor power from these two value that make the aircraft completely wing borne at the required transition time and rotor tilt angle.

4. Weight sizing:

The purpose of this section is to introduce a technique to obtain the first estimate of the MTOW (or all-up weight) for an aircraft before it is designed and built. the value for the MTOW is not final and must be revised in the later design phases. The result of this step may have up to about 20% inaccuracies, since it is not based on its own aircraft data. But the calculation relies on other aircraft data with similar configuration and mission. Thus, we adopt past history as the major source of information for the calculations in this step.

Since the accuracy of the result of this design step depends largely on the past history, one must be careful to utilize only aircraft data that are current, with aircraft that are similar in configuration and mission.

The most important weight in the design of an aircraft is the maximum allowable weight of the aircraft during take-off operation. This is also referred to as the all-up weight. The design MTOW or WTO is the total weight of an aircraft when it begins the mission for which it was designed. The maximum design take-off weight is not necessarily the same as the maximum nominal take-off weight, since some aircraft can be overloaded beyond design weight in an emergency situation, but will suffer a reduced performance and reduced stability. Unless specifically stated, MTOW is the design weight. It means every aircraft component (e.g., wing, tail) is designed to support this weight.

The aircraft weight is broken into several parts. Some parts are determined based on statistics, but some are calculated from performance equations.

The MTOW is broken into four elements:

1. Payload weight (W-PL).
2. propulsion system weight (W-PS).
3. Battery weight (W-Batt).
4. Empty weight (WE).

We do weight sizing by constructing two curves one from analytical equations and the other from statistical survey and the take-off weight is the point of intersection between the two curves.

1. Analytical equation:

Take-off weight is divided into payload weight, propulsion system weight, battery weight, and empty weight.

$$w_{TO} = w_{PL} + w_{Energy} + w_{Empty\,weight} + w_{PS} \qquad (20)$$

$$w_{TO} = w_{PL} + \left(\frac{w_{Energy}}{w_{TO}}\right) w_{TO} + \left(\frac{w_E}{w_{TO}}\right) w_{TO} + (\frac{w_{PS}}{w_{TO}}) w_{TO} \qquad (21)$$

$$W_{To} = \frac{w_{PL}}{1 - \overline{w}_{energy} - \overline{w}_{E-PS} - \overline{w}_{PS}} \qquad (22)$$

We need to find the relation between take-off weight (w_{TO}) and empty weight relative to take-off weight ($\overline{w}_{E-PS}$) according to the above equation the payload weight is given in design requirements, so we need to find battery weight relative to take-off weight ($\overline{w}_{energy}$),and find propulsion system weight relative to take-off weight ($\overline{w}_{PS}$).

1. Battery weight relative to take-off weight ($\overline{w}_{energy}$):

It calculated for each segment in UAV mission profile and then get the summation as follow:

$$\frac{W_{Batt}}{W_{TO}} \sum_{i=1}^{n_{seg}} \frac{W_{Batt,i}}{W_{TO}} \qquad (23)$$

Different UAV segments shown in the following table

And by the following equations we can find weight of each component of electric motor propulsion system relative to total take-off weight.

A. For initial design:

Table 1 initial weight sizing for propulsion system

Initial weight sizing
$\frac{W_{PP}}{W_{To}} = \frac{0.0014}{(PL)}$, for $P < 9[kW]$
$\frac{W_{prop}}{W_{To}} = 0.0161 K_{material} K_{prop} n_{prop}^{0.218} n_{blade}^{0.391} \sqrt{\frac{1}{\pi (DL) n_{prop} (PL)}}$
$\frac{W_{ESC}}{W_{To}} = \frac{0.0003}{(PL)}$

Where k_{prop} is factor depends on the engine type and its power and it has a value of (24 for turbopropswith P>1500hp, 31.92 for turboprops and piston engineswith P<1500hp,and 15 for engines with P<50hp).

Table 2Battery weight during different segments

Group type	Mission segment	$W_{Batt,i}/W_{TO}$
First (low energy) group	Conventional landing	0.001
	Vertical landing	0.002
	FW descent	0.001
Second (high energy) group	Conventional TO	$\frac{-\mu_f + \left(\mu_f + \frac{C^*}{C_{L.R}}\right) e^{\frac{\rho g C^* K_{ob} S_{TO}}{(WL)}}}{-\left[\cos(\theta_{STO}) + \mu_f \sin(\theta_{STO})\right] + \left[\cos(\theta_{STO}) + \mu_f \sin(\theta_{STO}) + \frac{C^*}{C_{L.R}} \sin(\theta_{STO})\right] e^{\frac{\rho g C^* K_{ob} S_{TO}}{(WL)}}} \frac{S_{TO}}{\eta_p \eta_{tr} e_{spc}}$
	FW climb	$\frac{V_y + \frac{1.155}{\sqrt{1/(4C_{Do}K)}} \sqrt{\frac{2(WL)}{\rho_o \sqrt{3C_{Do}/K}}}}{\eta_p \eta_{tr}} \frac{\Delta t}{e_{spc}}$
	Vertical TO	$\frac{F_{P+w}\left[(2-k_i)\frac{V_y}{2} + k_i \sqrt{\left(\frac{V_y}{2}\right)^2 + \frac{(DL)}{2\rho_o} F_{P+w}}\right] + \frac{\rho_o V_{Tip}^3}{(DL)}\left(\frac{\sigma C_d}{8}\right)}{\eta_{tr}} \frac{\Delta t}{e_{spc}}$
	Transition (low V)	$\frac{\left[\frac{k_i}{sin(\theta_{tr})} \sqrt{\frac{-V_{wb}^2}{2} + \sqrt{\left(\frac{V_{wb}^2}{2}\right)^2 + \left(\frac{(DL)}{2\rho sin(\theta_{tr})}\right)^2}}\right] + \left[\frac{\rho V_{Tip}^3}{(DL)}\left(\frac{\sigma C_d}{8}(1+4.6\mu^2)\right)\right] + \left[\frac{0.5\rho V_{wb}^3 C_{Do}}{(WL)} + \frac{2K(WL)}{\rho V_{wb}}\right] + \left[\frac{V_{wb}^2}{2g t_{tr}}\right]}{\eta_{tr} \sigma_{air}} \frac{t_{tr}}{e_{spc}}$
	Transition (high V)	$\frac{\left[\frac{k_i (DL)}{2\rho V_{wb} sin^2(\theta_{tr})}\right] + \left[\frac{\rho V_{Tip}^3}{(DL)}\left(\frac{\sigma C_d}{8}(1+4.6\mu^2)\right)\right] + \left[\frac{0.5\rho V_{wb}^3 C_{Do}}{(WL)} + \frac{2K(WL)}{\rho V_{wb}}\right] + \left[\frac{V_{wb}^2}{2g t_{tr}}\right]}{\eta_{tr} \sigma_{air}} \frac{t_{tr}}{e_{spc}}$
	FW range	$\frac{R}{\eta_{tr} \eta_p e_{spc} \sqrt{1/(4C_{Do}K)}}$
	FW endurance	$\frac{t}{\eta_{tr} \eta_p e_{spc}} \sqrt{\frac{2(WL)}{\rho \sqrt{3C_{Do}/K}}} \frac{1.155}{\sqrt{1/(4C_{Do}K)}}$
	Hover endurance	$\frac{t}{e_{spc}} \frac{1}{(FM)\eta_{tr}[1 - f_w (DL)/(WL)]^{\frac{3}{2}}} \sqrt{\frac{(DL)}{2\rho}}$

where

$$F_{P+w} = \frac{f_p V_y^2 + (WL)}{(WL) - f_w (DL)} \qquad (24)$$

is a term to account for the vertical drag force during the vertical climb and the rotor download force due to the rotor downwash impinging the wing and/or fuselage.

2. propulsion system weight relative to take-off weight ($\overline{w}_{PS}$):

For electric motor calculate the weight of each component separately and then substitute in the following equation

$$\frac{W_{PS}}{W_{TO}} = f_{installed}\left(\frac{W_{PP}}{W_{To}} + \frac{W_{prop}}{W_{To}}\right) + \frac{W_{ESC}}{W_{To}} \qquad (25)$$

$n_{prop}$is the number of propellers , n_{blade} is the number of blades per propeller, D_p is the propeller diameter in feet, n_{ENG} is the number of engines, $K_{material}$ is a material correction factor to account for propeller materialwith value of (1.3 for wooden propeller, 1 for plastic, 0.6 for compsite ones.

B. For detailed design:

Table 3 Detailed weight of propulsion system

Detailed weight calculations
$W_{PP} = 0.0099P^{0.7533}$, for $P < 9[kW]$
$W^*_{prop} = 0.0045K_{material}K_{prop}n^{0.218}_{prop}n^{0.391}_{blade}\left(D_pP\right)^{0.782}$
$W_{ESC} = 0.0031P^{0.7247}$

w^*_{prop} results in this equation in [lb], and D_p , P are in[ft], [hp],respectively.

2. statistical survey curve:

From survey of tri-tiltable rotor VTOL UAV we find statistical data about the values of take-off weight (w_{TO})and the values of empty weight relative to take-off weight ($\overline{w}_{E-PS}$).

Then by drawing two curves (one from analytical analysis and the other from statistical data) we find that the point of intersection of the two curves represent the total take-off weight as follow:

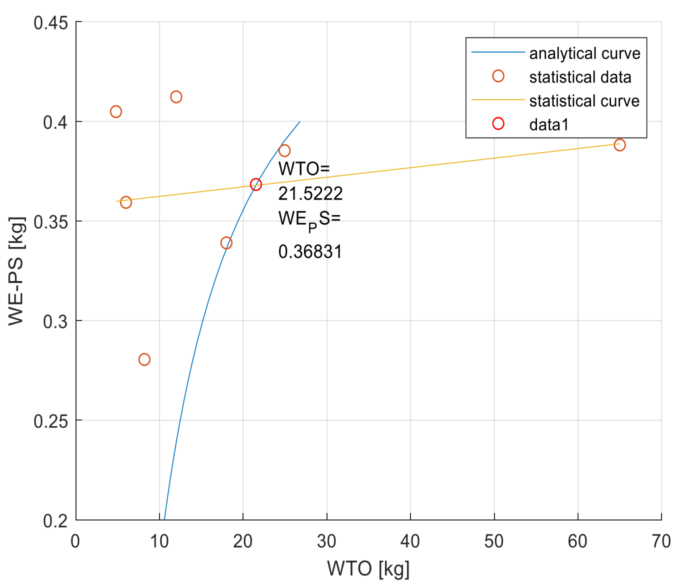


Fig. 5 determination of take-off weight

After determine the total take-off weight we can determine the wing area and power needed for fixed wing mode and hovering mode and the other parameter of sizing process as follow as shown:

Table 4 output of sizing process.

Parameter	Value	Parameter	Value
Take-off mass [kg]	21.5222	Empty weight [kg]	9.7404
Wing loading [N\m^2]	153.630	Battery weight [kg]	8.2814
Wing area[m^2]	1.3743	Power loading(FW) [N/W]	0.11794
Wing span [m]	3.5169	Power loading (VT) [N/W]	0.049899
Maximum lift coefficient [-]	1.5	Wing aspect ratio	9
Maximum power (FW) [W]	17901	Maximum power (VT) [W]	42312
Rotor disc loading [N\ m^2]	157.356		

After resizing process, the sizing parameters have small changes as follows:

Table 5 output of resizing process.

Parameter	Value	Parameter	Value
Take-off mass [kg]	23.59	Empty weight [kg]	11.3
Wing loading [N\m^2]	153.63	Battery weight [kg]	9.26
Wing area[m^2]	1.529	Power loading(FW) [N/W]	0.1166
Wing span [m]	3.6	Power loading (VT) [N/W]	0.0493
Maximum lift coefficient [-]	1.5	Wing aspect ratio	9
Maximum power (FW) [W]	2015	Maximum power (VT) [W]	4766
Rotor disc loading [N\ m^2]	160.23		

5. Airfoil selection:

The designer must also consider other requirements such as airworthiness, structural, manufacturability, and cost requirements. In general, the following are the criteria to select an airfoil for a wing with a collection of design requirements:

1. The airfoil with the highest maximum lift coefficient (Cl max).
2. The airfoil with the proper ideal or design lift coefficient (Cl-d or Cl-i) corresponding to both (Cl/Cd) max &(Cl^3/2/Cd) max.
3. The airfoil with the lowest minimum drag coefficient (Cd min).
4. The airfoil with the highest lift-to-drag ratio ((Cl/Cd) max).
5. The airfoil with the highest lift curve slope (Clαmax).
6. The airfoil with the lowest (closest to zero; negative or positive) pitching moment coefficient (Cm).
7. The proper stall quality in the stall region (the variation must be gentle, not sharp).
8. The airfoil must be structurally reinforceable. The airfoil should not be so thin that spars cannot be placed inside.
9. The airfoil must be such that the cross-section is manufacturable.
10. The cost requirements must be considered.
11. the typical wing (t/c) max should be less than 16%.
12. The airfoil with the highest lift-to-drag ratio ((Cl^3/2/Cd) max).

there is no unique airfoil that has the optimum values for all the above mentioned requirements. For example, you may find an airfoil that has the highest Cl max, but not the highest (Cl / Cd) max. In such cases, there must be compromise through a weighting process, since not all design requirements have the same importance.
The selected airfoil is (S2091-101-83) airfoil and has the following characteristic:

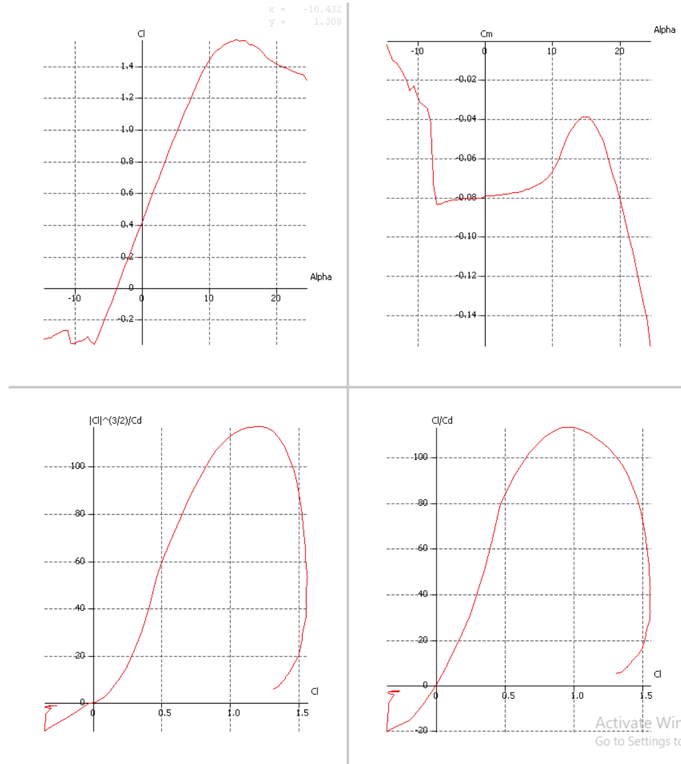


Fig. 9 airfoil characteristic curves

B. Design of UAV components

After completing sizing process and select airfoil of the wing we can now design UAV components.
Starting with wing main dimensions and design of its control surface (ailerons) then determine the main dimensions of horizontal and vertical tail and design of their control surfaces (elevator& rudder).and also we do an optimization for design of winglets to reduce induced drag and to increase the endurance of the UAV.
In wing design it’s very important to ensure that the lift distribution on wing closes to the optimum elliptic lift

Parameter	**Value**	**Parameter**	**Value**
Wing span [m]	3.68	Aileron area[m^2]	0.1151
Root chord [m]	0.44	Horizontal tail span[m]	1.02
Wing area[m^2]	1.528	Horizontal tail root chord[m]	0.32
Wing taper ratio(λ)	0.82	Horizontal tail taper ratio[(λ)	0.8
Mean aero-dynamic chord[m]	0.417	Horizontal tail sweep angle(Λ)[deg]	6.71
Wing sweep[deg]	0	Elevator span[m]	1.02
Aileron span [m]	1.104	Elevator area[m^2]	0.08874
vertical tail span[m]	0.4	vertical tail root chord[m]	0.32
vertical tail taper ratio (λ)	0.8	vertical tail sweep angle(Λ)[deg]	27.5
Rudder span[m]	0.4	Rudder area[m^2]	0.0348
Fuselage length[m]	1.12	Fuselage max diameter[m]	0.3

distribution.
The following Fig shows the lift distribution on the designed wing

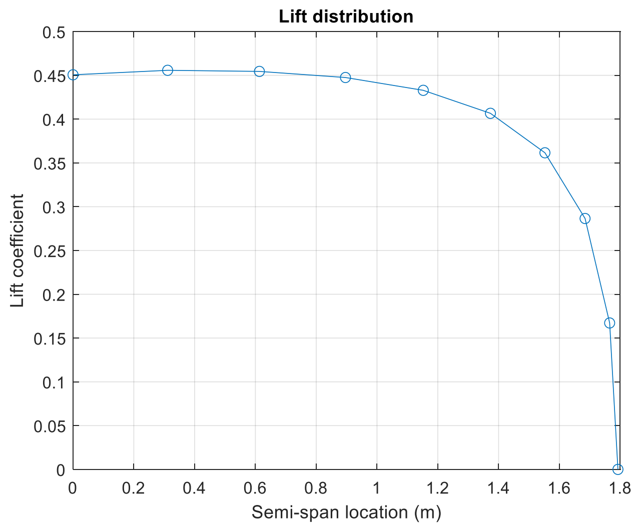


Fig. 10 lift distribution on wing

The detailed parameters of UAV components are shown in the following table

Table 6Dimension of different UAV components

C. Determination of UAV center of gravity and mass moment of inertia:

Center of gravity determined from the follwing relations:

$$X_{cg} = \frac{\sum_{i=1}^{n} W_i x_{cg_i}}{\sum_{i=1}^{n} W_i} = \frac{\sum_{i=1}^{n} m_i x_{cg_i}}{\sum_{i=1}^{n} m_i} \quad (26)$$

$$Y_{cg} = \frac{\sum_{i=1}^{n} W_i y_{cg_i}}{\sum_{i=1}^{n} W_i} = \frac{\sum_{i=1}^{n} m_i y_{cg_i}}{\sum_{i=1}^{n} m_i} \quad (27)$$

$$Z_{cg} = \frac{\sum_{i=1}^{n} W_i z_{cg_i}}{\sum_{i=1}^{n} W_i} = \frac{\sum_{i=1}^{n} m_i z_{cg_i}}{\sum_{i=1}^{n} m_i} \quad (28)$$

And mass moment of inertia is determined from the follwing equations:

$$I_{xx} = I_{xx_{CG}} + m(y^2 + z^2) \quad (29)$$

$$I_{yy} = I_{yy_{CG}} + m(x^2 + z^2) \quad (30)$$

$$I_{zz} = I_{zz_{CG}} + m(y^2 + x^2) \quad (31)$$

$$I_{xz} = I_{xz_{CG}} + mxz \quad (32)$$

The results are shown in the following table

Xcg	*0.151*	*Ixx*	*3.725*
Ycg	*0*	*Iyy*	*5.617*
Zcg	*-0.084*	*Izz*	*9.139*
		Ixz	*-0.177*

** Xcg measured from leading edge*

D. Longitudinal stability:

We consider a value of static margin (SM) about 15% from mean aerodynamic chord for this category of UAVS and the calculating the neutral point of UAV from the following equations:

$$SM = \frac{X_{np} - X_{cg}}{\bar{c}} \quad (33)$$

Then distribute the mass of each component along the aircraft body and do analysis by XFLR5 software that using vortex lattice method to determine the lift, drag, pitching moment of the UAV as shown.

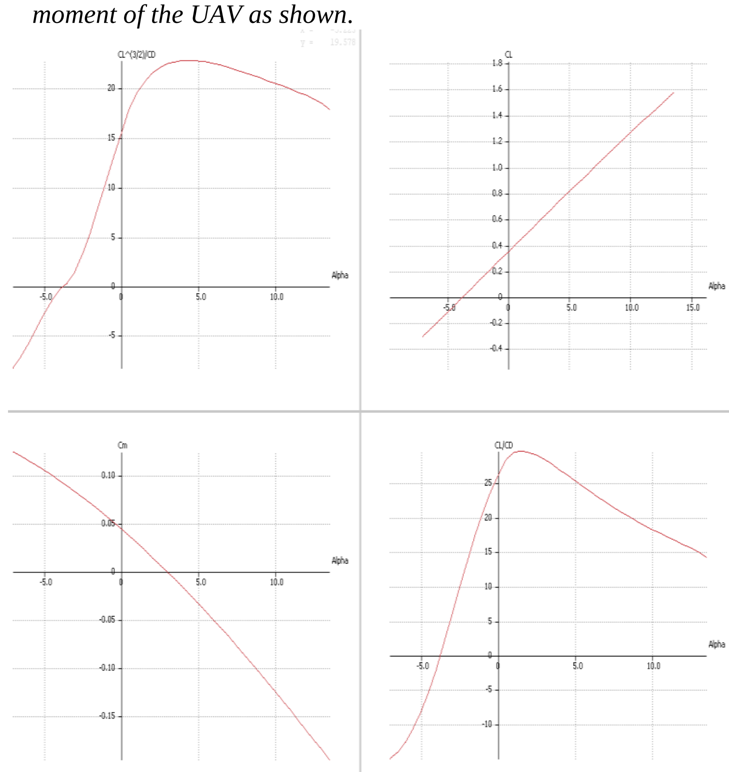


Fig. 11 UAV characteristic curves

E. propulsion system selection:

We use two front motor with tilting mechanism these motors must cover the required power for both fixed wing mode and hovering mode in vertical flight. The selected front motors are (Dual sky ECO 5322C) with propeller (18x10). And the rear motor must cover the required power in vertical flight. The selected rear motor is (Dual sky XM6360EA-11) with propeller (20x12). These motors are supplied by lithium polymer batteries that must cover the required endurance.

The selected batteries are (Lipo31000 6S 22.2v Battery pack) and lithium-ion battery (6S 30000mAh solid state LI-ion battery).

F. Manufacturing of UAV:

The design of wing depends on simply spar semi-monocoque structure with main and auxiliary from carbon fiber and ribs from ply wood and balsa wood. Similarly, for horizontal and vertical tail. the fuselage design depends also on spar semi-monocoque structure as it contains bulk heads from ply and balsa wood and main boom from carbon fiber connecting bulkheads to each other. The detailed design is shown

Fig. 12 Internal structure of wing

Fig. 13 internal structure for fuselage

The following 3-D view will illustrate the full dimension of different component of the UAV.as shown in Fig.14

CONCLUSION

This paper presents a methodology for sizing FW-VTOL electric UAV. The sizing converts the design requirements into basic parameters of the designed UAV. The performance constraints are developed for the rotorcraft, transition, and the fixed-wing flight modes to develop the matching curve charts. This design chart enables selecting the optimum preliminary design parameters through obtaining the adequate values of the power, wing, and rotor disc loadings that are needed to satisfy the design requirements in the three flying modes. This methodology allowed to avoid the unneeded excess power and consequently reduced the aircraft weight and cost.

The actual components of the propulsion system and battery are selected based on the results of the initial sizing. Aircraft resizing is performed to update the parameters of the aircraft using information about designed/selected parts. FW constrain analysis, electric propulsion sizing, mission battery calculations, total mass calculations, and geometry analysis, are combined into FW-VTOL electric UAV integrated analysis. The developed sizing methodology provides a useful tool for aircraft designers at both early and detailed design stages. Future work will include the manufacturing and flight testing of the designed transitional UAV.

ACKNOWLEDGMENT

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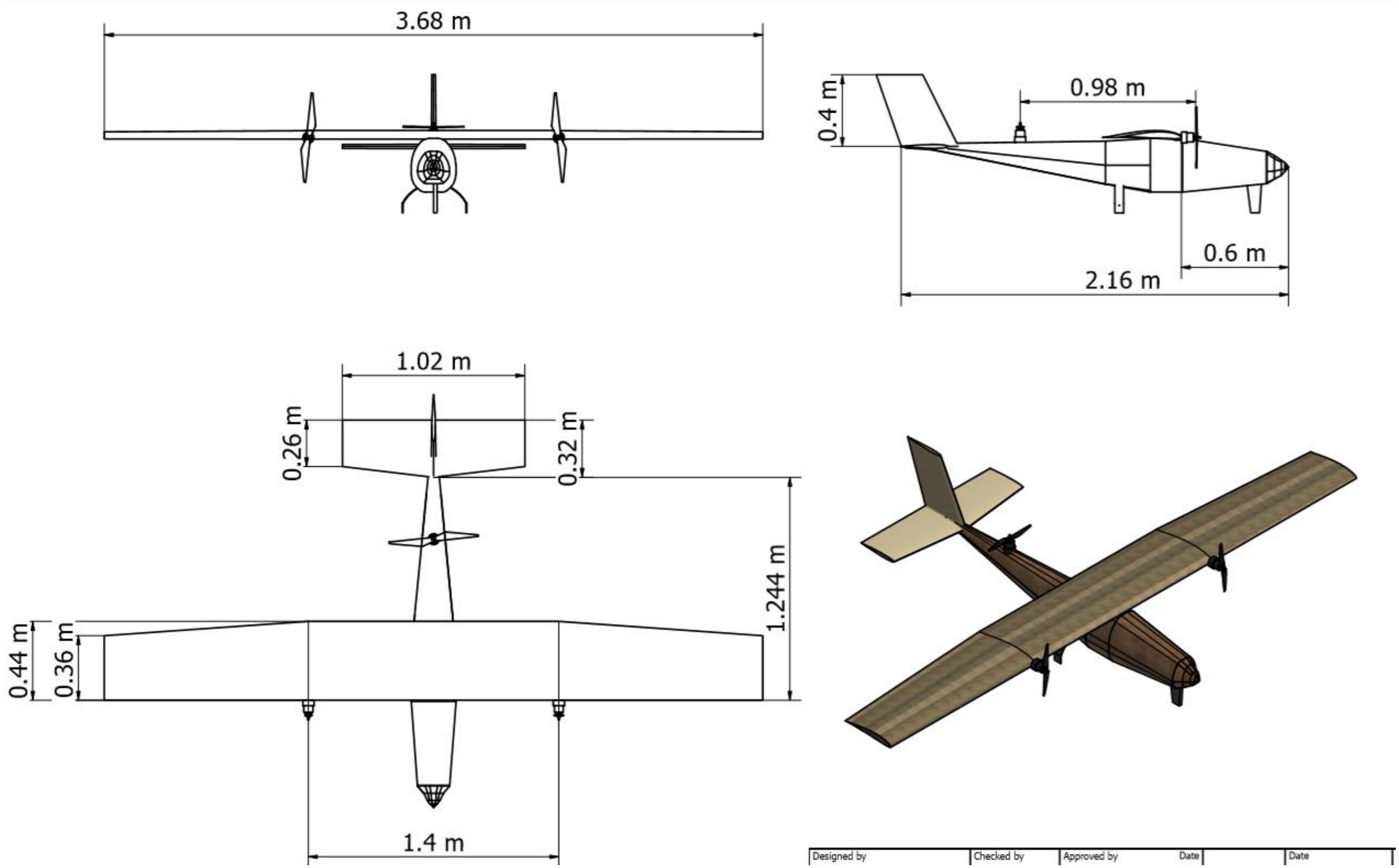

Fig. 14 3-D view of UAV