7. ANKARA INTERNATIONAL AEROSPACE CONFERENCE
11-13 September 2013 - METU, Ankara TURKEY

AIAC-2013-012

DESIGN FABRICATION OF HAND LAUNCHED UAV

Adel Ali Kurban[1], Gamal Mohamed Ashawesh[2,]
Ahmed Assidiq Elhuni[3], Arij Mohammad Thabet[3], Hussein Farj Bizan[3]
University of Tripoli
Tripoli, Libya

ABSTRACT

Unmanned Air Vehicles have been the subject of increased attention for the past decade because of their unique potential in both military and civilian applications. These include surveillance, search and rescue.

Desiging Aircraft is a complex multi-disciplinary task and teaching and by placeing the student in the real word problem, in order to achieve an overall perspective of the aircraft design process. The approach chosen at University of Tripoli is intended to create the right balance between theory and practice.

This Paper summarizes the designing project carried out by team from the Aeronautical Engineering Department at University of Tripoli, Libya. The team was formed to design, build, test and fly a hand-launched electrical power UAV.

Hand-launched UAV was designed to perform low altitude surveillance. The design process was started by a set of requirements; these requirements were used to size, and estimate the aerodynamic parameters, then the design process was followed by selection of electrical motor and battery capacity. The construction process was started after many design iteration cycles and it was made from composite materials. Static tests for the wing were carried out to make sure that the wing structure could carry the applied loads without any structure failure.

Key Word: UAV, Design, Fabrication.

INTRODUCTION

The design process for new aircraft must have established requirement to serve as the focused goal for final design [Torenbeek E,1982]. This paper present the development of Radio controlled (R/C) hand launched Electrical powered [Kurban A. et al. 2011].

This paper describes the various aspects involved in the design of a electrical powered unmanned aerial vehicle (UAV) to complete a successful hand-launch, the weight of the aircraft must be limited to a reasonable throwing weight and the dimensions of the throwing surface limited to fit in the user's hand and would have a total mass of 4 kg. The aircraft had a conventional geometry of main wing, and back tail. Due to hand launch the aircraft to land safely on its belly by designing the bottom of the aircraft to absorb impact energy

The aircraft's main structure was based on carbon fiber, which have the highest strength to weight ratio. The main wing had a carbon fiber top and bottom flange spar as well. This makes it both light and simple to assemble. The payload compartment was built as a separate structure, which was not part of the aircraft body. This specific design allowed easy aircraft loading with its third mission payloads.

[1]Assc Prof. in a Department of Aeronautical Engineering, Email: kurban@aerodept.edu.ly
[2] Prof. in a Department of Aeronautical Engineering, Email: gashawesh@aerodept.edu.ly
[3] postgraduate studuent in a Department of Aeeronautical Engineering,Email :A.Elhuni@aerodept.edu.ly

Project objectives:

The project has number of objectives

1. Enabling the student to use the different knowledge and skills acquired during their academic studies.
2. Experiencing the aircraft design, development, manufacturing , and testing process.

Design requirements:

One of the first and often most important step in the aircraft design process are to define all of the requirements that will be imposed on the design. Typically, the majority of these requirements is directly requested by the customer and is not negotiable. Some of these parameters could include a minimum range, cruise speed, or payload capacity. The specific combination of requirements depends on the design mission.

In order to help guide the development of the UAV, a set of specification were established. The aircraft design requirements are summarized in table 1 below.

Table 1 Design requirements

Take-off and landing	Maximum Takeoff weight 4 Kg
Stalling speed	not exceed 9 m/s
Endurance	Not less than30 min
Wing span	maximum 2 m
Propulsion	electric motor
Airframe	robust construction, easy inexpensive manufacturing cost.
Payloads	Maximum Pay load weight 1 Kg (interchangeable CCD cameras, on board recording device, and sensors)
Navigation	GPS / autopilot
Operation	Radio controlled flight one man operable
Mission altitude	150 m from the sea level

UAV DESIGN

Performance Matching

The initial weight estimation was made base on historical statistical data, and use of weight fraction, [Roskam,1999].

The purpose of the performance matching is to select the appropriate wing loading (W/S), power loading (W/P), and maximum lift Coefficient, C_{Lmax} based on the performance requirements specified in table 1. The performance requirements imposed on the performance matching are stalling speed of 9 m/s and maximum cruising speed of 17 m/s, the performance-matching plot shown in figure 1 was generated using the previous requirements.

Based on four constraints, the power loading and wing loading for different maximum lift coefficients and aspect ratio were used to determine a wing loading under which all constraints would be met. Constraint analysis plot (Figure 1). The chosen values of wing loading, power loading, and maximum lift coefficient are:

$\left(\frac{\mathrm{W}}{\mathrm{S}}\right)_{\mathrm{TO}} = 74\ \mathrm{N/m^2}$, $\left(\frac{\mathrm{W}}{\mathrm{P}}\right)_{\mathrm{TO}} = 0.095\ \mathrm{N/watt}$, and C_{LMAXL}=1.5 and $\mathrm{C_{LMAXTO}}$ =1.3

Wing loading, power loading, and maximum lift coefficients were chosen as the result of several iterations trading airfoil selection, and performance requirements.

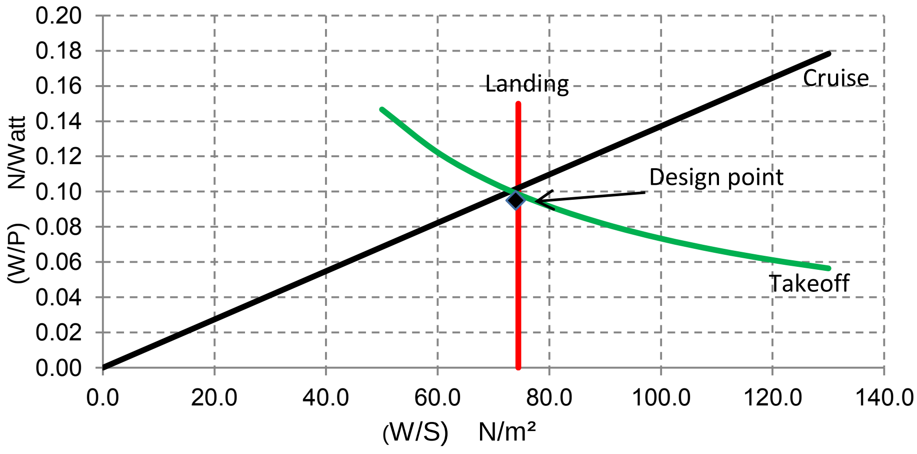


Figure 1: Constraint analysis

Configuration and Layout

Four basic airframe configurations were considered in the design. Every design was judged by five main characteristics: Empty system weight, manufacturability, stability and control, and aerodynamic performance. After the general configuration of the airplane was determined, few decisions about the configurations of specific parts were made.

Payload area geometrical and location

It was desirable to design the payload compartment in such a way that the CG location will not vary with different loadings of the airplane. Carrying the payload inside an external pod enables flexibility and modularity of the design by allowing future changes of the CG location and compartment size. A round sectioned cylindrical shape was chosen for the pod mainly because of manufacturing considerations.

Wing geometry and location

Hand-Launch of the aircraft was the main reason for mounting the wing as high as possible on the fuselage in order to provide a comfortable grip of the UAV body near the CG while keeping a good clearance between the launchers head and the wing. Locating the wing above the CG also improved the lateral stability creating a pendulum effect. A tapered outer wing section shape was chosen. This is considerably easier to manufacture though improves the induced drag.

Tail

Three tail arrangements were considered from several aspects: it is simple, easy to build, and usually provides adequate stability and control with a light structure weight.

Propeller location selection

Propeller location was mostly affected by aerodynamic and power considerations. However, other aspects like the safety of the person who launches the UAV affected the design as well. Folding Propeller where selected to offers better ground clearance during landing

Preliminary Aircraft Arrangement

The fuselage, wing, and tail designs developed were combined to create a preliminary arrangement of the aircraft. This arrangement was then used to locate each of the weight components as shown in figure 2. This was then used to tabulate the fuselage station, water line, and buttock line of the center of gravity of each weight component. The centers of gravity for the fuselage, wing, engine cowl, and empennage were then estimated using CAD drawing and data taken from [Raymer D, 1991]. These values were used to determine the center of gravity for the vehicle

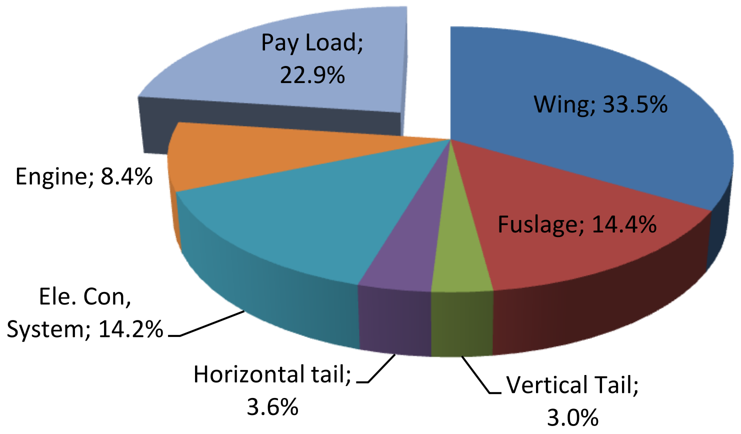


Figure 2: Graphical representation of component weight breakdown

Three-View Drawing

Detailed drawings are a key part of any design process; it has great benefit in weight and balance, stability and aerodynamic calculations. These drawings of the internal and external component arrangements, and all necessary modification were made throughout the design process.

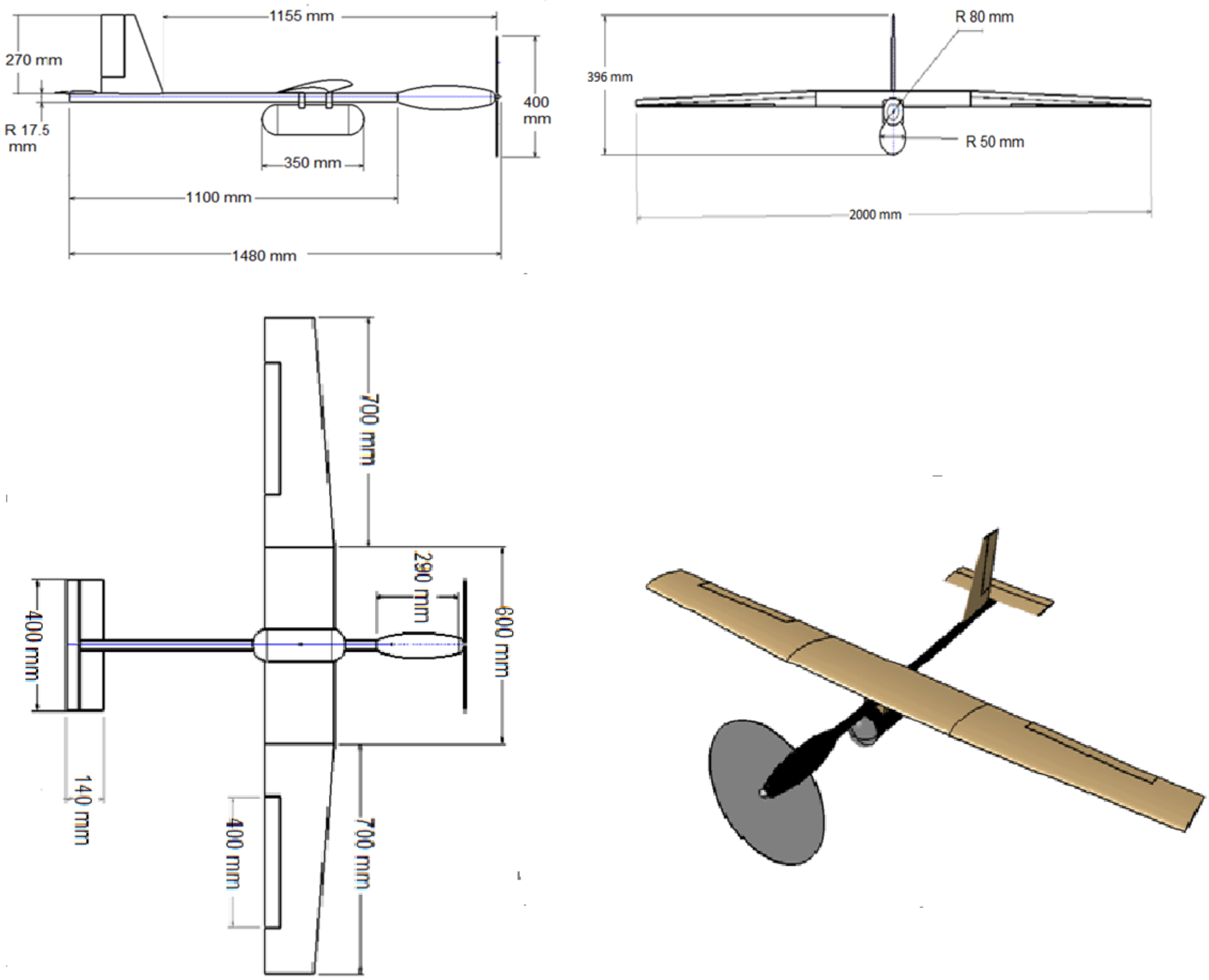


Figure 3: Three view layout Final configuration

Estimation of the Drag Performance

The parasite drag was estimated for the optimized UAV, the estimation was based on the components build up method [Anderson, J.D, Jr., 1999]. More detailed information on components build up method can be found in, [Raymer, D., 1991, Roskam, J., 1983]. Two types of wing drag were considered, parasite drag and induced drag. Then, the wing drag was added to other components (Fuselage, tail, wing etc.) to obtain the total parasite drag coefficients of the UAV. The induced drag was also estimated for the wing to complete the drag polar relation, which was found to be

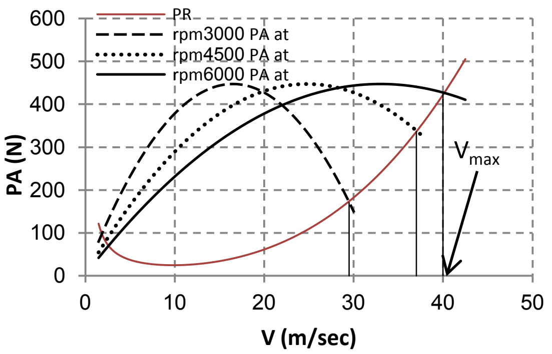
$$C_D = 0.022968 + 0.046\, C_L^{\,2}$$

The flight performance of the aircraft is described by point performance of the vehicle. The thrust available was calculated from the Excel spredsheet. The thrust required curve was calculated using the drage coefficent equation.

The maximum velocity of the aircraft occurs at the point when the thrust required is equal to the thrust available.

The power required and available at three (3) different RPM were plotted versus velocity, the excess power obtained from these plots were used to determine the rate of climb for specified RPM and maximum velocity at sea level.

This data was used to find the maximum velocity for each mission. The maximum velocity for maximum engine RPM was found to be 40 m/s and maximum rate of climb as shown in figure 4 and figure 5.



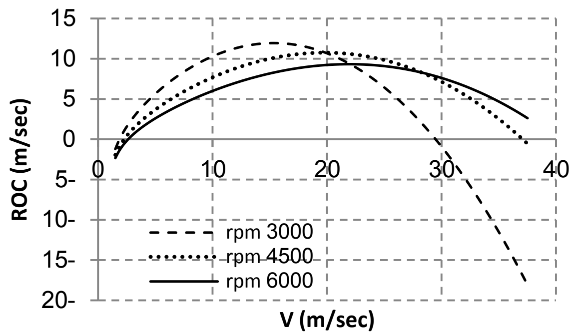


Figure 4: Power Available & Required Versus velocity V

Figure 5: Rate of Climb (ROC) versus velocity (V)

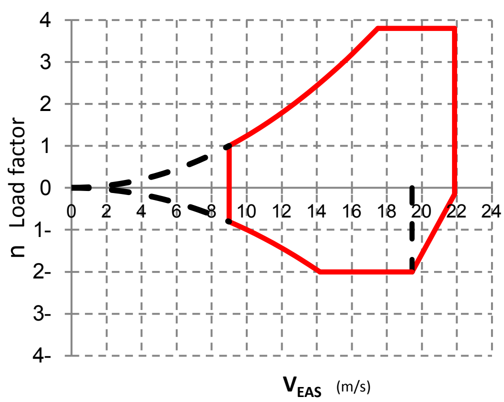
Structure

The regulations for UAV structural design are not well established in comparison to manned vehicles; thus, UAV design often depends on the manufacturers' experiences.

In this work, the structural design and analysis results of a relatively UAV made of composite materials are presented. The main objective was to verify its structural performance by tests and a simple theoretical modeling. Static strength test for the wing was performed by sand bag loading.

Maneuver and Gust Envelopes

The v-n diagram (Figure 6) was calculated based on JAR-VLA requirement [Anon, 2011], the load factor for cruise speed and diving speed were determined, these result shown in (Table 1).



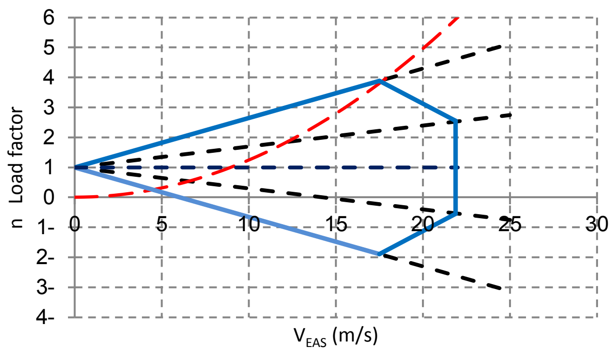


Figure 6: Maneuver and Gust Envelopes versus Velocity

Table 1: Maneuver and Gust Load Factor for cruise and diving Speed

Speed	Normal Load Factor	Maneuver	Gust
V_C	Ng (+)	3.8	3.89
	Ng (-)	-1.5	-1.89
V_D	Ng (+)	3.8	2.54
	Ng (-)	-0.14	-0.54843

Wing Shear Force and Bending Moment

Schrenk's approximate method [Howe D., 2004] was adopted and since there is twist, the basic load distribution is zero, the additional loading distribution corresponding to own overall lift coefficient of unity, the span wise air load distribution for wing was calculated for different loads corresponding to v-n maneuver and gust cases for aircraft mass of 3.5 kg at sea level.

The wing was divided into small elements in span wise direction (0.05 m) width, close integration of air loads was preformed to calculate the shear force distribution and then the bending moment, the result were presented in figure 7 and figure 8, respectively.

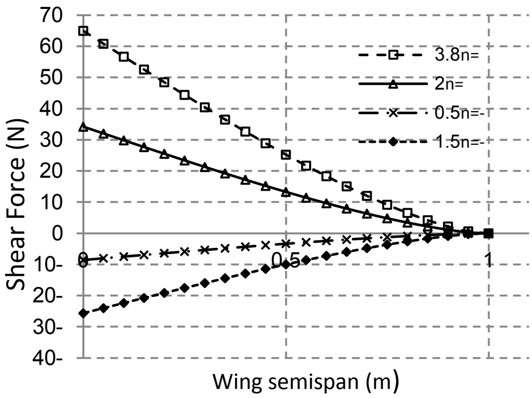


Figure 7: shear force along semi span

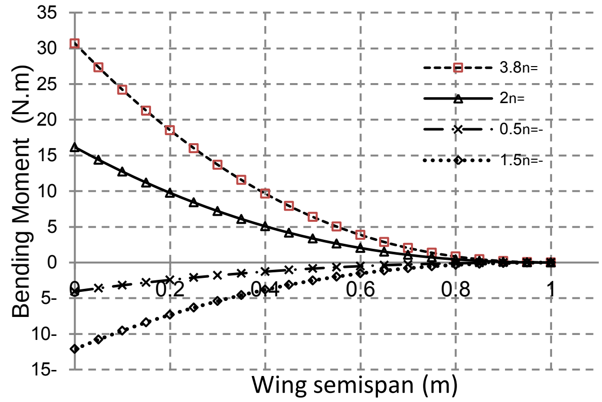


Figure 8: bending moment along semi span

Wing Structure Testing

The maximum load in the order of 2 gs compared to the limit design load of 3.8 gs for which the structure design is considered. It is therefore rather safe to fly your aircraft before you perform the structural wing tests and possibly destroy your aircraft before you take to the air.

Calculation of the vertical deflection of the wing requires knowing the spanwise bending stiffness distribution EI(y) along the primary axis of loading. For a wing made of a uniform solid material, the modulus E is a simple scaling factor. The moment of inertia of the airfoil cross-sections about the bending axis x (bending inertia), is related only to the airfoil shape given by the upper and lower surfaces, both the area A and the total bending inertia I are the integrated contributions of all the infinitesimal rectangular sections.

The stiffness of the beam in bending is calculated from the equivalent flexural rigidity, (EI) eq which represented by Equation

$$(EI)_{eq} = \frac{E_f {b_t}^3}{6} + \frac{E_c {b_c}^3}{12} + \frac{E_f b t d^2}{2}$$

$$(EI)_{eq} = 23.92 \ \ N.m^2$$

The first and second terms describe the stiffness of the two face sheets and the core while the third term adds the stiffness of the faces about the centre of the beam.

Young’s modulus of wing can be obtained by estimate the equivalent moment of inertia

$$\mathrm{I} = \frac{\mathrm{c\,h}^3}{12} = 98751.95\mathrm{mm}^4$$

$$\mathrm{E} = \frac{(\mathrm{EI})_{\mathrm{eq}}}{\mathrm{I}} = 242150000\mathrm{N/m}^2$$

Where, h is the average height of the wing section and c is the wing root chord, assuming that the wing as cantilever beam subjected to uniform transverse spanwise distributed load, this cause spanwise deformation due to shear force & bending moment.

The deflection equation is derived as follows:

$$\delta = \frac{b^2 \times M_0}{E\, I_0} = \frac{N\, W_{UAV}}{E\, I_0} \times \frac{b^3}{96} \times \frac{1+2\lambda}{1+\lambda}$$

The team conducted bending tests on the wing, one simulating static loading, and one with distributed loading on the whole wing spar, in order to simulate the lift force.
Eight dial gauges were placed in wing edges. At first, an initial load was applied to the wing to eliminate any initial setting, and then the measuring instruments were zeroed.
The shot bags were loaded, starting from inboard to outboard as indicated in the models. Readings were taken after loading at n=1 and 2. The readings were taken again for unloading wing.

The arrangement of these measuring instruments is shown in figure 9 and the comparison between theoretical and test deflection readings are presented in figures10.

Figure 9: Wing Test

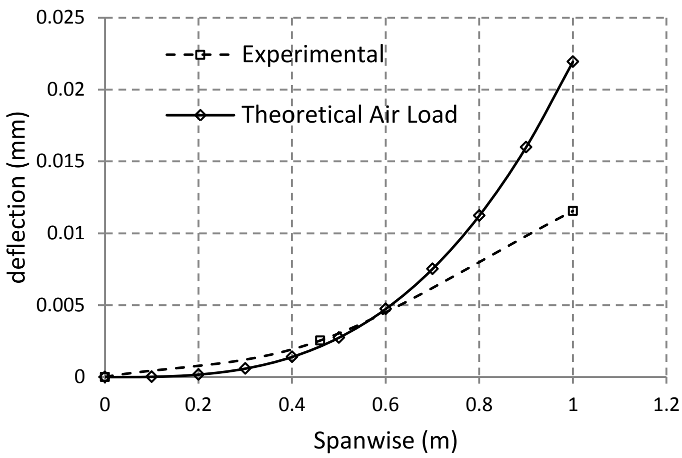


Figure 10: Semi-span Deflection at (n=2)

FABRICATION

After the detailed design was completed, the team started to build the airplane. Balsa, foam, and carbon fiber were considered as main materials for the construction of the aircraft, the team examined several different methods and processes that would provide the desired characteristics for each major component (wing, empennage, and fuselage). Selection of the best-suited manufacturing technique is depending on the component type and shape.

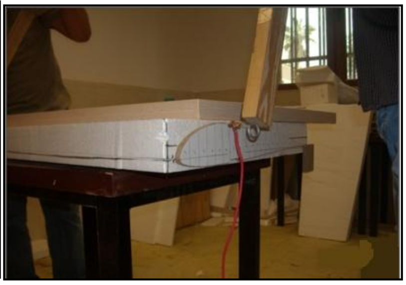
Wing and tail unit

The mold less composite construction technique was used to build the wing, horizontal tail and vertical tail. This technique allows the builder to construct a part by forming a core material to a desired shape and then laminating the reinforcement material to the shaped piece to make up the final part.
The core structure made of polystyrene foam of density 15 kg/ m^3, foam core usually remains as part of the structures to aid in maintaining rigidity and shape.
A wing core is typically fabricated by making templates. Using nails, which are pushed through holes in the sides of the foam, weigh from sand bags used as downward force to hold down the foam on a flat table. Two people are needed to guide the hot wire cutter over each template. Each template is divided into an equal number of spaces and each space is numbered. One-person calls out the number were his wire is located on the template and the other person makes certain that his end is at the same number.
The Balsa and Composite Hybrid skin carries the main loads of the structure at inner wing section and the balsa sheet only at outer wing. This method is common for manufacturing wings.

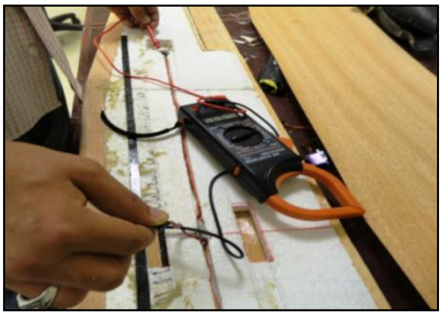
Horizontal and Vertical stabilizer

Stabilizer was constructed in a very similar manner to the outer parts of the main wing. The vertical stabilizer was made from one layers of very fine balsa wood sheet and core structure made of polystyrene foam.


Wing Template


Hot Weir Foam Cutting


Wing All Cuts Completed


Wing servo motor wiring


Wet layup trimming preparation


Entire wing color binding

Figure 11 Wing part Making Steps

Fuselage nose and pod

A foam model of the fuselage is created by hand, which is then used to create a gypsum mold. This mold can then be used to create a strong fiberglass female mold. This mold can then be used to made fuselage nose part and under wing container pod the steps are shown in figure 12.

The pod was designed to be tolerant to impact, but must be light enough. A carbon and Kevlar shell was made, with foam core to support payloads, a wooden block was hand carved on a lathe. Then female fiberglass mold was manufactured. The pod was made longer than designed, and was cut out to the correct size later.


Coated Foam block


Nose Plug


Nose Mold reverse side

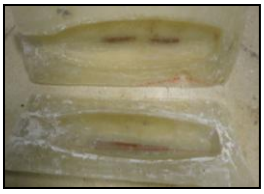

Nose Mold

Figure 12: Mold Making Steps


Carbon Laminate Wet Layup


Complete Nose

Figure 13 Fuselage Nose part Making Steps

Figure14: UAV ready for Flight test

Conclusion

Hand launched electrical powered UAV was successfully designed fabricated and tested. During the process, several changes had been made to ensure that the mission profile is met.

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