Design of an Aircraft for the BMFA Payload Challenge (Weight)

Report

For

A. White and N. Revill

[British Model Flying Association]

University of Strathclyde Group A

Reference Number: W14 Team A

May 2016

Abstract

An aircraft was to be designed to meet the rules of the British Model Flying Association University and Schools Payload Challenge 3 (Weight). This requires an aircraft carrying a water payload of up to 4kg to take off within a distance of 61m, complete a predefined circuit and land successfully. The competing aircraft will be judged on their payload to empty aircraft mass ratio. The main limitations of the design were the requirement to use a specific motor (the E-flight Power 10) and the simulation of sensory equipment using a 150mm diameter polystyrene ball. The initial target of the design was to carry the maximum 4kg payload with an aircraft of empty mass 1kg which could takeoff within the specified distance and was stable and controllable in all configurations. In order to achieve this it was decided to package the payload within the wings so as to reduce the structural mass of the fuselage.

The design process for the aircraft involved obtaining a baseline design which was capable of taking off within the prescribed distance and was stable in flight and then optimising it with the aim of reducing the empty aircraft mass and improving the stability characteristics such that they were unaffected by the addition of the payload. Once this had been achieved the aircraft was constructed. This proved to be a challenging process during which several changes were made to the design for ease of manufacture and to improve its strength. These changes significantly increased the mass of the aircraft.

The final aircraft design is capable of carrying the full 4kg payload however the empty aircraft mass target was not achieved. The final mass is 1.98kg which whilst failing to meet the target set was felt to be acceptable given that the difficulty in packaging the payload in the wings was higher than expected at the beginning of the design process. Carrying the maximum payload the aircraft is 11.6% stable and capable of taking off within 42.8m. It has a stall speed of 10.6m/s and a maximum speed of 18m/s. Whilst this is a little low it is felt that it is still an acceptable range of speeds which will allow it to be flown carrying 4kg. Overall therefore whilst the mass target for the aircraft was not met it was nonetheless a successful, innovative design which met the other targets set at the beginning of the design phase and successfully allowed the payload to be carried within the wings as originally envisioned.

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1.0 Nomenclature

Abbreviation	Meaning	Symbol	Meaning
BMFA	British Model Flying Association	C_{M0}	Pitching Moment Coefficient at Aerodynamic Centre
HFD	Horizontal Fuselage Datum		
UTS	Ultimate Tensile Strength		
AOD	Aft of Datum (Wing Leading Edge)		
FEA	Finite Element Analysis		

2.0 Introduction

The BMFA University and Schools Payload Challenge tasks a group of students to design a model aircraft to a given specification and then compete against other similar groups with the aircraft they have produced. In this case the challenge selected by the group was the weight challenge. This requires the aircraft to carry a removable payload of up to 4kg and possess as large a payload to empty aircraft mass ratio as possible. Three runs will be attempted with a maximum payload for each run of 0, 2 and 4 kg respectively. The aircraft must be powered by a specified electric motor (the E-flight Power 10) and carry a removable polystyrene ball designed to simulate sensory equipment. The centre of the ball must therefore be located a minimum of 400mm from the motor to simulate negating the effect of electrical interference and vibrations[1]. There are restrictions upon the components which can be placed in the 'field of regard' of this ball - a 60 degree cone below the ball taken from the points where the HFD (a horizontal line passing through the aircraft nose) crosses the ball. In terms of performance the aircraft must take off within a 61m take-off run and be stable and capable of being trimmed for level flight for all flight speeds and all possible payloads.

This design first required a philosophy to be established and concepts to be generated to meet this philosophy. Once the optimum concept was selected an initial design was developed which would allow the aircraft to take off and be stable in flight. Detail design and optimisation was carried out to produce the final design. Finally the aircraft was constructed with alterations made to the design as appropriate in order to make this process easier or to correct oversights or unforeseen problems with the design. All of the stages discussed above will be covered by this report along with an overview of the aircraft's final design and a discussion of its performance. Finally the final aircraft design will be compared to the design targets set at the beginning of the project.

3.0 Design Philosophy

Once the weight challenge had been chosen it was necessary to select a design philosophy which would allow the payload to empty mass ratio to be maximised. Firstly it was decided that the maximum possible payload (4kg) would be targeted with the objective being to produce the lightest possible aircraft which could carry this payload. It had been advised that an empty mass of 1kg would be achievable and this was reinforced by studying previous competition data. As such it was thought that in a worst case scenario the mass would be no more than 1.5kg which would still provide a high payload to mass ratio of 2.67 (the winning team in 2015 achieved a ratio of 2.76 [2]) should the full 4kg be carried. Secondly it was decided to attempt to locate the payload within the aircraft wings. This would result in the fuselage being required to support a much smaller load and as such greatly reduce the structure and weight required. In addition it would reduce the in-flight moment experienced by the wings whilst making use of the existing structure within the wings to support the mass of the payload. The design was largely driven by the takeoff, particularly during the initial design phase, since this was the stage of the flight at which performance was most marginal and the aircraft most likely to fail.

4.0 Design Process

4.1 Concepts

After some provisional calculations of the volume of the wing approximating it as a 2m span, 0.25m chord and 0.03m thickness box and taking into account the loss of available volume due to the solid sections of the wing such as the ribs, it was felt that there would be sufficient volume in the wings for the payload. A provisional calculation of the moment generated by the 4kg payload estimated that the maximum stress experienced by the spar in this scenario would be around 12.5MPa (assuming a load factor of 5 during landing). It was felt that this would allow a relatively light material to be used for the spar whilst still providing the necessary strength. As a result it was decided that packaging the payload within the wings was feasible.

Two concepts were generated for the positioning of the ball within the aircraft. The first was to locate the propeller at the front of the aircraft and locate the ball 400mm behind it, the second was to switch these positions. It was concluded that packaging the propeller in this way would be very difficult and severely affect its performance. As such it was decided to accept the greater difficulty in packaging other components around the ball's field of regard and place the ball around 400mm behind the front of the aircraft.

It was also necessary to select the position of the wings relative to the fuselage. It was decided that the weight compromises which would have to be made in terms of fuselage structure in order to mount the wings below the fuselage were not justified by the relatively small improvement to the drag coefficient during the take-off run and would limit the positions available for the wing due to the field of regard of the ball. As such it was decided to mount the wing on top of the fuselage.

It was decided initially that attaching the undercarriage above the fuselage would not be desirable as it would require a larger and hence heavier undercarriage. Having two separate legs would minimise the size of the undercarriage and so this concept was chosen at this stage to minimise weight.

With the exception of the undercarriage, alterations to which will be discussed in Section 4.3 these concepts were realised in the final design of the aircraft and can be viewed in Drawings 1 and 2.

4.2 Initial Design

The main goal for this phase was to produce a design which would be capable of taking off within the prescribed distance of 61m. For a short take-off distance the wing dimensions would preferably be as large as possible and the wing profile would provide as high a maximum lift coefficient as possible. Other conflicting demands also affected the initial design of the wings however, namely the need to save weight by having a smaller wing and the need to select a wing profile which would simultaneously allow the maximum payload to fit into the wing and provide good performance in flight at low Reynold's Numbers. Hysteresis loops from stalling were to be avoided since the aircraft would not be flying high enough to recover from such a stall.

It was estimated that the Reynold's Number at which the aircraft would be operating during flight would be in the range of 150,000-250,000. In order to prepare for the likely and worst case scenarios the performance at Reynold's Numbers of 100,000 and 200,000 were studied for a selection of wing profiles. As the maximum payload for the BMFA challenge was to be 4kg of water the wings would require at least 4 litres of usable volume inside of them. After an initial comparison the FX 63-1370 was selected as the wing profile for the aircraft. This was one of the thinner wing profiles studied, however it was initially believed that the payload would fit into this wing. The FX 63-1370 wing profile also performs very well in low Reynold's Number conditions and has a high lift coefficient when compared to the other aerofoils that have a regular flight behaviour at these low Reynold's Numbers[3].

The provisional dimensions chosen for the wing were a span of 2.2m and a chord of 0.25m. This allowed the necessary volume to carry the payload and the large wing area required for take-off whilst also minimising the size of the wing and hence the structure and weight necessary. A C-section spar with squares of height and breadth 2.5cm connected by a shear web was initially chosen to support the wings and payload as this section is the most efficient shape for the spar. It was thought that these initial dimensions would be a good basis to optimise from as when stress calculations were performed a large safety factor of around four was present. Cyparis was chosen as it had a higher UTS compared to other common construction woods such as Birch or Balsa wood whilst also having a relatively low density[4].

Another major factor in the take-off was the drag coefficient of the aircraft. A MatLAB script was written to calculate this value taking into account skin friction, form, interference and wing profile drag. This was only an estimation of the true drag coefficient of the aircraft but it gave a good enough approximation to be used in the take-off equation. The initial design of the fuselage was heavily based upon minimising the drag that it would produce. Since the ball (diameter 150mm) had to be contained within the fuselage this dictated its maximum diameter which was initially estimated at 180mm to allow for the ball and some structure around it. It was decided to target an ellipsoidal fuselage which would be circular in shape when viewed from the front and elliptical when viewed from the side with the widest point being where the ball is contained. This would allow drag to be minimised and cut back on the structure and weight necessary for the fuselage. The feasibility of manufacturing this shape had not been considered and resulted in changes which will be discussed in Section 4.3.

The final element which had a significant effect on the take-off run of the aircraft was the thrust curve. Using the wind tunnel data provided for a range of propellers[5] the thrust curves for each were plotted using Microsoft Excel. It was found that even using props of a relatively similar size the take-off distance could be increased by up to 10-15m if the wrong option was chosen. As such, after testing the thrust curves of all available props using the takeoff equation it was decided that the 12x6 propeller was the best choice as it provided the most thrust. It was also felt that any associated increase in mass compared to a smaller propeller would be negligible, especially when the advantage provided by the extra thrust generated was taken into account. After the use of E-Calc and discussion with the technician it was felt that the battery available would be able to provide the necessary power for this propeller whilst also providing enough endurance (4.1 minutes) for the competition.

The rigging angle of the wing and angle at which the propeller was mounted could be optimised for a small improvement in take-off distance. It was decided that these values should be optimised for the take-off run as such optimisation was a key part of the design philosophy. At this stage no values were selected since the design of the aircraft was expected to change significantly and hence they would have to be re-optimised once the design was finalised. However the optimised values were calculated for each design which was created and used to help in assessing its feasibility.

Finally at this stage a rough estimate of the mass and position of each component within the aircraft was made in order to estimate the centre of gravity. It was decided to use a flat plate for the tailplane to obtain initial values for stability. Using the estimated values it was found that the aircraft was unstable in all configurations and thus it was clear that work would have to be done on this area during the next phase of the design.

4.3 Design Refinement

It was found that one of the major factors affecting the stability of the aircraft was the position of the wing relative to the nose of the aircraft. This was because the wing was a major contributor to the total mass of the aircraft and hence the position of its centre of gravity and also since the position of the aerodynamic centre affected the position of the neutral point. In addition the wing was one of the few components whose position could be significantly altered without changing the design concept of the aircraft. As such the wing leading edge was moved back from its original estimated position of around 250mm aft of the nose to 300mm aft of the nose in order to obtain a baseline stable configuration.

The size of the tailplane was also increased to increase the tail volume coefficient and hence move the neutral point backwards on the aircraft improving the stability. The tailplane was kept as a flat plate for simplicity of manufacture. The tail volume coefficient could also have been increased simply by moving the tailplane further back relative to the wing. This was not done since it would increase the length of the aircraft and hence the structural mass required to connect the tailplane to the fuselage. Whilst increasing the size of the tailplane also necessitated an increase in the mass it was felt that this would be less severe since balsa wood would be used to construct the tailplane and this material was very light. By increasing the tailplane from its poorly estimated initial dimensions - 0.1m chord, 0.3m span - to more realistic dimensions - 0.25m chord, 0.5m span - the tail volume coefficient was increased into the advised range of 0.3-0.7 [6].

Whilst this initial study was useful in obtaining an approximately stable layout there were several problems with it. The first was that all of the masses used were rough estimates which in some cases proved to be fairly inaccurate. As such lab testing of the motor, battery, speed controller, servos and of undercarriage designs from previous years was necessary in order to obtain more realistic values for their mass. The second issue was that the variation of the centre of gravity and hence stability when adding the payload was quite large; the stability ranging from 8-33%. It was preferable to have the stability within the range of 10-20% for all configurations since this would give a margin of error to keep the aircraft stable but also ensure that the aircraft was not sluggish to control. In order to ensure that the stability did not vary unacceptably with payload mass it was necessary to locate the centre of gravity of the payload as close to the centre of gravity of the empty aircraft as possible. A target value of empty aircraft centre of gravity of 70mm AOD (the wing leading edge was chosen as the datum) was established as the payload could be located in this region. In this configuration the neutral point lay at 120mm AOD and hence a static margin of 20% was achieved. With some minor alterations (moving the centre of gravity further aft of the datum slightly) these positions remained largely constant throughout the remainder of the design process. A concern caused by the payload was that all containers used on a given flight would need to be filled completely otherwise the payload and hence its centre of gravity would move during flight. As such it was necessary to have multiple containers in order that flights with different payload masses could be undertaken and it was therefore necessary to produce a design which was stable for three different configurations.

As the design progressed and the structure necessary for the wing became clearer it became obvious that the FX 63-1370 was not thick enough for the payload and its containers to fit into the wing. This meant that a thicker aerofoil had to be selected. More accurate calculations of the maximum speed of the aircraft and of the cruising speed were made and it was found that the Reynold's Number across the wing would be higher than initially estimated, around 250,000, which allowed a wider choice of aerofoils. The Clark Y aerofoils were identified as they were simpler than most to construct and well behaved at the operating Reynold's Numbers. The Clark YM-15 profile[7] was selected as it was the thinnest and hence lightest profile which could feasibly contain the payload.

It was found that 4 cylindrical containers of length 1.02m (leaving space in the middle of the wing to connect to the fuselage) and diameter 0.036m would hold a total of 4.15 litres and would fit into the Clark YM-15 profile however on closer inspection the edges of these cylinders were too close to the edge of wing profile and the ribs would not be feasible to manufacture in reality. Whilst an arrangement of elliptical containers with similar volume solved these problems since they made better use of the space in the wing they were judged to be too difficult to manufacture in the time available. Furthermore, the centre of gravity for this elliptical arrangement was too far rearward.

A final concept which was tested was that of creating a hollow spar with the payload container located within. This appeared to be a viable concept initially since it reduced the mass of the wing by removing the need for separate structure to support the payload containers.

This design was also found to have insurmountable issues. Firstly due to the location of the spar at the thickest point of the wing the centre of gravity of the payload was too far aft of the wing leading edge for a stable configuration to be reached. Secondly the size of spar required for a four litre container was very large. Even when various measures were taken to reduce the mass of the spar by removing material at points where it was not required this resulted in a much higher spar mass than alternative designs.

It therefore became apparent that the 0.25m chord length was not large enough to contain the payload. The chord were increased to 0.3m allowing two cylinders with a diameter of 37mm to carry 2 litres each and fit into the Clark YM-15 profile with enough room to allow adequate space for the spar and the thickness of the containers and the ribs.

At this time the control of the aircraft was also considered more carefully. A value of 0.4 was chosen for the elevator to chord ratio since it provided an acceptable lift curve slope of 3.7/rad without the elevators occupying too high a percentage of the tailplane volume resulting in issues with the functionality of the elevators. Secondly the C_{M0} value for the wing was required. Since this value was not specifically provided for the Clark YM-15 it had to be calculated using the data provided. This involved making use of the fact that when the sum of the forces normal to the wing chord line are equal to zero the moment coefficient is the same for all points on the wing. This value is hence equal to C_{M0} since the value of pitching moment coefficient is constant at the aerodynamic centre of the wing[9]. When the provided data for lift and drag of the wing was studied this resulted in a value for C_{M0} in the region of between -0.07 and -0.08; due to the lack of data points provided this value had to be interpolated. It was decided to use a C_{M0} value of -0.08 since this provided a worst case scenario for the moment required of the tailplane. Once these values had been calculated it was clear that, using the initial configuration, the range of elevator angles to trim from stall speed to maximum speed was far too large. However once the centre of gravity had been located at the target value of 70mm AOD these problems were largely solved.

The magnitude of the tailplane lift coefficient required to rotate the aircraft during take-off had to be lower than the maximum lift coefficient of the tailplane (0.8) otherwise the tailplane would not be able to produce enough lift for rotation at take-off. To ensure that this was the case the rear undercarriage was moved forwards relative to the tailplane, increasing the moment arm of the tailplane about the undercarriage and hence reducing the lift coefficient required to rotate. Once this change had been made the lift coefficient to rotate was around 0.8. Ideally this value would be lower to take into account the potential separation of the flow with elevator deflection, however this was still superior to the previous value and could be optimised during the final phase of design.

The main factor controlling the aileron volume coefficient was the space available within the wings. To maximise the available chord length inside of the wings it was decided that the ailerons would cover the majority of the wingspan, leaving 20cm in the centre to mount to the fuselage. The given range for the aileron volume coefficient was 0.3-0.44. The final dimensions of the two ailerons were chosen as 0.993m span and 0.083m chord for each aileron giving a volume coefficient of 0.0342.

The governing factor controlling the size of the fin was the weight so the fin was designed to be as small as possible. The fin chord is 0.125m and the height is 0.15m. Using the dimensions and placement of the fin gave fin volume coefficient as 0.018. The rudder was chosen to be 0.03m by 0.13m. Given the rudder's placement and its dimensions the rudder volume coefficient was calculated to be 0.051. Both of these values were within the specified limits[6].

Since the payload was stored in the wings and not the fuselage this allowed significant weight savings to be made as the required structural mass was minimal. The only structure necessary was that which was holding all the essential components and that giving the fuselage its shape.

Whilst the preferred fuselage design was an elliptical shape as discussed in Section 4.2, owing to a lack of resources and time it was concluded after consultation with the lab technician that this was unfeasible to manufacture. Another concept was devised in which the fuselage was split into two parts connected by a boom; the essential components would be housed behind the propeller, with this connected by a forward boom to the ball container situated underneath the wing. Upon more detailed FEA it was found that the stresses experienced by the boom would cause all lightweight materials available to rupture. As a lightweight fuselage was a primary design goal, this idea was discarded.

This led to a simple design of four connected cuboids being selected, the width and height of the fuselage would vary for each cuboid; increasing from front to back which acts as a taper to reduce drag. Whilst this option still created more drag than other designs due to the rear of the fuselage ending abruptly after the polystyrene ball, this was offset by the savings in weight. Hatches were simple to create as there were only straight edges involved and this allowed quick and easy removal of the ball fairing. The largest cuboid is located at the rear of the fuselage, contains the ball and mounts to the wing. Balsa wood was selected as the fuselage material in order to save weight. The design can be viewed in Drawings 6 and 7.

At this stage of the design greater consideration was also given to the undercarriage. It was felt that the initial intention to mount the undercarriage to the fuselage would require too much structure to be added and undo much of the work done to reduce mass in this area. Instead it was decided to mount the undercarriage to the wing since it was designed to withstand far greater stresses.

The material chosen for the undercarriage was a woven epoxy carbon fibre due to its high stiffness and strength and low weight. The undercarriage geometry was modelled using FEA and analysed assuming that the load factor during the landing is five. The maximum tensile and compressive stresses were found to be 352 MPa and -306 MPa respectively which are acceptable when compared to the orthotropic compressive stress limits of 805 MPa and -509 MPa as found in the ANSYS Workbench Engineering Data section.

4.4 Design Finalisation

After the decision had been made to increase the wing chord and use cylindrical containers for the payload there was some freedom in the positioning of the containers for the purpose of stability. As such the containers were positioned either side of the spar which could support their mass. With both full containers present this resulted in a payload centre of mass at approximately 83mm AOD, giving 11.6% stability for the maximum payload run. When only one container was filled this would inevitably move the centre of gravity of the payload and as such it was unavoidable that a small change in stability would occur on the 2kg run, however due to the close proximity of the container to the empty aircraft centre of gravity this was minimal. It was decided to use the forward container on this run since it meant that the stability would remain within the 10-20% range.

The changes made to the design of the fuselage discussed in Section 4.3 resulted in the initial estimate of its centre of gravity being too far rearwards. Since the fuselage made a significant contribution to the total aircraft mass this had a large effect on its stability. In order to maintain the necessary stability characteristics the wing was also moved forward. Locating the wing leading edge 230mm aft of the nose ensured that the no payload centre of gravity lay at 85mm AOD and hence ensured stability for the no payload (11%) and maximum payload (11.6%) runs were very similar. These positions can be viewed on Drawing 2. An issue posed by this positioning of the wing was that the largest cube within the fuselage design did not start until after the leading edge of the wing which gave potential issues with mounting the wing. It was felt that by creating a plate to mount the wing onto this could be solved.

An additional positive outcome of moving the wing and hence the centre of gravity forward in this way was an increase in the tail arm value. This was beneficial since it brought the magnitude of the lift coefficient to rotate below the maximum lift coefficient of the tailplane to around 0.34 and ensured that it would be feasible to rotate in this configuration. In addition the new range of elevator angles was preferable to the previous range which was a little too large. At this point the aircraft design was largely finalised and hence the wing setting angle could be selected. As discussed above this was optimised to provide the shortest takeoff run and was for this reason set with the chord line at 8º relative to the HFD. The optimum propeller mounting angle was found to be too small to justify the extra difficulty involved in achieving this angle so was not optimised for takeoff.

In total 13 ribs were chosen across the span of the wing produced using balsa wood in order to minimise weight. The payload itself was to be contained in cylindrical containers which were to be purchased from external sources. As specified in the rules, the payload is removable and this is achieved by having slots on the outer edge of the wings so the cylinders can be eased into the wing. The containers were intended to be supported via a bracket system connected to the spar by adhesive. These brackets were intended to be composed of an aluminium alloy due to the impressive strength to weight ratio. As will be discussed in Section 5.0 this proved to be unfeasible.

As discussed in Section 3.0, it was decided that the payload would be stored in the wings primarily to reduce the wing stressing during flight since this weight will act in the opposite direction to the lift force on the wing. This means that the maximum stress in the wing would be when it is not in flight itself but during landing where the load factor was estimated to be 5. If the wing were to exceed this critical load factor in flight however, this would cause a wing loading for which the plane was not designed. This load factor corresponds to a banking angle of 78°. Since this is a very high angle it is safe to assume that the wing load factor will never reach this value and so the root moment of the wing will never be greater during flight than when it is landing. It was possible to achieve the required stressing using a C-section spar design created using 5x5mm cyparis and a lite-ply shear web. This design resulted in a maximum stress of 11.9MPa. Cyparis could be used as discussed in section 4.2 since its yield strength is 25.5MPa [10] and as such the design has a safety factor of 2.1.

5.0 Construction

The horizontal and vertical stabilisers, rudder and elevator were produced using 5mm balsa, with the grain lying parallel to the longest dimension to increase strength. Small triangular gussets were glued into the corners to strengthen the joints of each component. The leading edges of the stabilisers and control surfaces were sanded to produce curved edges and the trailing edges of all four were filed to produce shallow triangles. This improved the airflow over the tailplane and allowed free movement of the control surfaces. This was achieved through cutting small slots into each component so that lightweight hinges could connect the rudder and elevator to the stabilisers. Before these were glued in place the entire tailplane was covered in a thin plastic skin which was heat shrunk to size, this reduced the skin friction and increased the maximum force which could be produced by the elevator and rudder.

Re-evaluation of the calculation of rudder volume coefficient resulted in the conclusion that it had been misinterpreted and that the size of the rudder would have to be increased. This resulted in a rudder of dimensions 125mm x 150mm (a triangle was cut into the bottom of this shape in order to allow the elevator to move freely since the rudder was positioned above it) with a volume coefficient of 0.065.

The tailplane is connected to the fuselage by a carbon fibre boom. A carbon fibre tube of external diameter 20mm and thickness 1mm was purchased to create the rear boom. The tailplane was glued to a triangular balsa platform which was attached to the boom using epoxy resin. The elevator and rudder servos were attached to small ply plates which were also mounted to this balsa platform. Thin rods were then used to connect them to the control surfaces. The servo wires ran down the inside of the tube which was cut to a length of 525mm. Rather than cut a slot for the wires to exit the boom it was decided that it would be simpler to leave the end open and route the wires out of the end.

It was decided that the vertical members of the fuselage would be attached to the horizontal members using square finger joints to create the ‘sides’ of each box with a face joint used to connect the third member at each corner to create the forward and rear faces of each box. These methods were chosen after testing the strength of a variety of mock joints and selecting those which best combined strength and ease of construction. It should be noted that as with the tailplane the grain of all pieces of balsa used ran in parallel with the longest dimension of the member. Thin strips of balsa were added running between the corners of the boxes to support the skin and create a more streamlined shape. There was a concern that the joints in the fuselage would not be strong enough to withstand forces from the front undercarriage during landing or the vibrations from the motor so a glass fibre strip of approximately 15mm in length was added to each corner allowing them to be strengthened with minimal weight penalty.

Several other changes were made to fuselage in order to make it more robust and increase the ease of access to the components stored inside it, although these added mass to the otherwise lightweight structure. The rear box was increased in height from 170mm to 175mm to allow some clearance for the ball container, the hatch for this was added to the bottom of the box. A strip of marine birch ply was added on top of this box for the rear boom and the rear undercarriage to be mounted onto without bolting straight into the balsa of the fuselage. This was also reinforced with glass fibre strips. A balsa insert was mounted to the top member of the rear face of this box which was designed to cradle the rear boom and give the tailplane its setting angle of -4º relative to the HFD. This setting angle was chosen to trim the aircraft at 80% of its maximum speed at maximum payload (14m/s).

Hatches were added to the bottom of the second largest and top of the second smallest boxes to allow access to the battery and speed controller respectively. The hatch positions were chosen to maximise ease of access since the wing and front undercarriage would make access in any other position difficult. The material used for the motor and front undercarriage mount was changed as even with reinforced joints balsa would not be strong enough to withstand the loads from the front undercarriage during landing and motor vibrations. It was replaced by a 6mm thick solid plate of marine birch ply. The undercarriage was attached to the ply plate on three sides of the motor to distribute the impact of landing.

The rear undercarriage was created using glass fibre rather than carbon fibre since this made it possible to manufacture it in house. The mould was made from two pieces of foam creating a rhombus which was 300mm high and 340mm wide at base. The first undercarriage created exhibited weakness at the corners when tested so a second was created to rectify this fault which proved to be much sturdier and withstood the loading which it was predicted to be exposed to during landing. The undercarriage assembly including the wheels weighed approximately 100g more than predicted, this was due to the change of material and the fact that the mass of the wheels had not been included in this estimate.

The first section of the wing to be constructed was the ribs. They were laser cut from 3mm lite-ply. Ply was necessary since it had been deemed unfeasible to create a bracket to support the payload containers, it was decided to instead support the containers using the ribs hence a stronger material was required. Although this added significant weight to the design it was the only solution which was feasible to manufacture and as such this weight increase was unavoidable. Of the 13 ribs 3 were 'full length' while the remaining 10 were 'half-length' to allow the remainder of the rib section to be used to construct the ailerons. The first set of ribs were cut to the shape of the Clark YM-15 aerofoil with 5x5mm squares cut to allow space for the spar and with two holes of 37mm diameter for the water containers. However these turned out to be too small to easily fit the containers and whilst space had been cut to allow for the leading and trailing edges there was no room for the d-box or the false leading edge therefore revised ribs were required. For the revised set of ribs the container holes were altered so that they were slightly elliptical in shape with major axis 40mm and minor axis 38mm. This allowed room for the containers to be rotated to fit if they were not perfectly circular in shape.

The spar was created using 5x5mm cyparis as planned. Since no 2.2m lengths of cyparis were available it had to be created by joining two pieces together. Since these joints were potentially weaker than the rest of the spar it was necessary to locate them as far from the wing root (where the highest stresses were located) as possible. The trailing edge, leading edge and false leading edge were created from 10mm balsa. It was cut and filed by hand to match the profile of the aerofoil, again two pieces had to be joined as no 2.2m lengths were available. When these were glued into place and filing of the leading edge was completed some small gaps existed between the leading edge and d-box which were filled in with polyfiller and then filed down again to obtain the required profile. Lastly the shear web was laser cut from 3mm lite-ply. The thickness was reduced slightly by sanding to leave a little more room for the payload containers to pass freely through the wing. Lite-ply was chosen for the shear web because the payload being carried in the wings meant that the extra strength was required.

Balsa strips were glued into the gaps between the ‘half-length’ ribs on the trailing edge to allow hinge slots to be cut for the attachment of the ailerons. The ailerons were created using the rear sections of the ‘half-length’ ribs and glued onto web sections made from 5mm balsa. Due to an accumulation of minor errors in the construction process the aileron leading edges did not exactly match the trailing edge of the wing itself and as such a larger than desired gap exists between the two. This will negatively impact the effectiveness of the ailerons. The aileron servos were mounted between two ribs halfway along the span of each aileron. A balsa frame was constructed around the servos which allowed the skin to be attached smoothly.

Due the size and shape of the wing the skin was attached in sections. Once the skin was in place, the shape of the aerofoil was altered as the skin was tight over the ribs but did not follow the profile of the wing between the ribs. This would not have been a problem if there were more ribs in place throughout the length of the wing. Due to the tight time scale and cost of altering the design, it was decided that there was no possible way of increasing the number of ribs so the decrease in the performance of the wing would have to be accepted. Were the design to be repeated the lack of ribs would be a key area to address. Another key fault in the wing design was that a lack of knowledge about the process of constructing the wing led to its mass being greatly underestimated however much of the extra mass added was necessary for the wing to function correctly so this increase in mass had to be accepted.

The payload containers were crucial to the wing construction as the entire wing might have to be redesigned if the containers themselves could not be manufactured to the required size. After extensive research, and with the relatively small budget in mind, it was decided that the containers would be made in house; enquiries into pre-made containers were given quotes of around £50 (which would have used a large proportion of the available budget). Following further research and testing with various materials, it was concluded that the best readily available material to construct these was the plastic used to make the 2 litre drink bottles. Subsequently, a test model was built that proved to be relatively successful- there was a slight seepage of water from the joins of the plastic, however this was easily resolved by taking more care when applying sealant to the joints.

The finalised 2.2m containers consist of dissected 2 litre drink bottles (each of the same brand for consistent thickness of plastic), a thin layer of sealant applied around the joins of the plastic to prevent the problem of water seepage, a foam stopper sealed in at one end and a removable foam stopper at the other to allow water intake/outtake. To further reduce the risk of leakage at the end of the removable stopper, a film was placed over the end of the container. A potential issue arose when during testing it was made apparent that, owing to the snug fit of the containers in the ribs, small parts of sealant were being eroded away through contact and allowing water to leak. This was simply solved by re-applying a thin layer of sealant to the leak.

As the connection between the wings, undercarriage and fuselage would experience very high forces – when landing in particular – it was necessary to compromise the weight saving philosophy further and use two marine birch plywood plates for this connection. One of these plates was attached directly to the top of the largest fuselage box whilst the larger plate was attached to the bottom of the wing. To save weight without seriously weakening this plate a sequence of triangles were cut out one of which served the secondary purpose of allowing the wiring for the servos to pass from the fuselage into the wing itself. Also attached to this plate was a vertical piece of ply used to set the wing rigging angle to 8° by mounting dowels to it and to the leading edge of the wing.

The fuselage was attached to the rear undercarriage underneath the other components. The rear boom was also attached to the underside of the rear undercarriage. Due to the fact that the tailplane and tail boom were set to an angle of -4° the angle that the undercarriage rested at was altered and as a result the large ply plate that sat on top of the undercarriage was also set at an angle of -4°. To counter this issue a small piece of balsa wood was attached to the rear boom to correct the angle at which the undercarriage sat, thus ensuring the wing rigging angle was correctly set at 8°.

The components were mounted using three M6 bolts. The first of these went through the undercarriage, rear boom and the ply plate on the fuselage. The other two bolts were placed through the wing's ply plate, the undercarriage and then the fuselage plate. This layout can be viewed in Drawings 1 and 2.

The motor was attached to the motor mount using four M3 bolts. The front undercarriage was then placed on the motor mount and secured using six plastic clips which were screwed to the motor mount. As part of the assembly before flight the battery and speed controller will be secured in the front of the fuselage just behind the motor mount. The receiver is located at the joint between the two largest boxes, allowing easy access to the wiring if necessary through the various hatches installed in the fuselage. The wiring was tested during the calibration of the servos to ensure that the servos and motor operated as intended.

As discussed above the highest stresses experienced by the wing were expected to be when it was landing with the payload contained within the wing. The wing could not be tested in this condition but was tested with the payload in the wing which was calculated to be equivalent to a load factor of 4 and was found to hold the payload comfortably. However the leading edge of the wing was found to droop down due to the payload being contained forward of the mounting points. This meant that the setting angles were not as intended. In addition the wing rested on the fuselage in this configuration and it was felt that the fuselage would not withstand a landing with the additional stresses that this would impart. To combat this the remaining section of the carbon fibre tube was cut into two. These rods were attached to the undercarriage such that the wing could not deflect down but instead rested on these rods.

6.0 Final Design

Once all phases of the design had been completed the final configuration of the aircraft could be defined. A non-tapered Clark YM-15 wing has been used with a span of 2.2m and chord of 0.3m. The wing setting angle is optimised for takeoff with the chord line at 8º to the HFD. The stresses in the wing are sustained by a C-section spar made using cyparis and a lite-ply shear web. The payload will be held in two plastic containers within the wing which are supported by the plywood ribs. These containers are cylindrical with a diameter of 37mm and will each contain 2 litres of water hence allowing the aircraft to carry the maximum payload of 4kg, they will be slotted into elliptical holes in the ribs of major axis 40mm and minor axis 38mm. This will result in a small change in the stability characteristics of the aircraft during the 2kg payload run since only one container will be in use. The aircraft will be 11% stable with no payload, 11.6% stable with the full 4kg payload on board and 16.3% stable when carrying the 2kg payload. The tailplane is a flat plate with an elevator to chord ratio of 0.4 and a setting angle of -4º relative to the HFD. The range of elevator angles to control the aircraft is 6.6º to -5º for the maximum payload case and 6.6º to -4.5º for the no payload configuration.

The aircraft is powered by the specified electric motor and uses a 12x6 propeller such that the thrust produced is maximised. The fuselage consists of a series of boxes of increasing size with the motor, battery and speed controller packaged within the first two of these. The mandatory polystyrene ball is contained within the final box, upon which the wing is also mounted. Finally the undercarriage is mounted to the wing so as to decrease the load experienced by the fuselage. The structural mass of the aircraft is 1.98kg. The predicted payload to empty mass ratio of the aircraft is therefore 2.02.

7.0 Performance

Once the aircraft design had been finalised, its final performance characteristics at maximum payload could be calculated and analysed in more detail. The maximum payload stall speed was found to be 10.6m/s. In order to find the maximum speed and maximum rate of climb of the aircraft the programme MotoCalc was used to calculate the thrust produced at different speeds. The power required (drag x speed) curve was then plotted along with the power available (thrust x speed) curve as shown in Figure 1. The point at which these curves crossed such that power required was equal to power available was the maximum speed of the aircraft[11], this had to be approximated from the graph as the data obtained using MotoCalc did not allow this point to be plotted directly. ECalc was therefore also used to compare the values obtained and was found to agree with the results obtained using Figure 1.

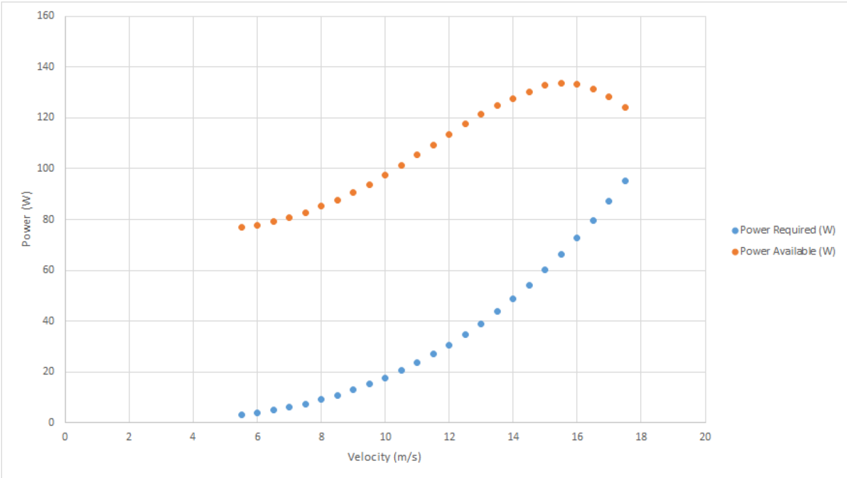

Figure 1: Power Required and Power Available for the Aircraft Carrying Maximum Payload

The maximum speed of the aircraft was extrapolated to be 18 m/s. This gave a range of speeds at which the aircraft could fly of around 7.5m/s for the maximum payload. This was a quite low (ideally it would be between 10 and 15m/s) however it was also not prohibitively small and it was felt that any changes made to improve this would require fundamental alterations of the design. As such it was felt that this range of speeds was acceptable since it should still allow the aircraft to be controlled and would not provide too small a window of speeds in which the aircraft must be maintained. The maximum rate of climb occurred when the difference between power required for straight and level flight and available power was at a maximum and hence the power available to climb was also maximised. This occurred at 12.5m/s where the excess power was equal to 83W. This gave a maximum rate of climb of 1.4m/s. The distance to clear 11m was found to be 40.2m (this assumes climbing at 12.5m/s when the maximum excess power of 83W is available and is hence a best case scenario). Whilst it is clear from this value and the maximum rate of climb that the aircraft does not climb particularly quickly it is also clear that for the particular purpose of this aircraft a high rate of climb is not necessary since the aircraft must only take off, fly a circuit and land.

Knowing the battery capacity rating, 2200mAh, and the current drawn at the speed where there is the most unused power, 16A, the range for the aircraft could be calculated. The maximum range of the aircraft was calculated to be 4.97 km. The endurance for the aircraft was found to be 4.1 minutes. These values for range and endurance were acceptable for the application of the aircraft. Such a high maximum range implied that it would not be a concern during the competition since it was a far greater distance than the aircraft would be attempting to cover whilst the maximum endurance was felt to give enough time to complete the necessary flight. The final take-off distance was found to be 42.8m, assuming the calculated final parasite drag coefficient of 0.05. This gave ample distance to take-off within the prescribed distance of 61m even in the event that the drag coefficient was a little higher than calculated which was thought to be likely given the difficulty in calculating it accurately.

The final predicted payload/mass ratio is 2.02. Whilst this did not reach the stated goal of a ratio of 4 this was in the range of those achieved at the competition the previous year[2] (although it should be noted that previous years were restricted by a maximum wingspan of 2m which no longer applies this year) and as such was thought to be an acceptable final ratio given the difficulties encountered during the design and build process.

8.0 Conclusion

The initial decision to locate the payload within the wings proved to be a very challenging concept to implement successfully. The fuselage mass was reduced to some extent by the need for minimal structure but compromises had to be made due to the need to maintain the aerodynamic performance and ensure that the aircraft was strong enough to withstand the loads which it was expected to experience. As a result the initial target mass of 1kg could not be achieved. Improvements could have been made to these results if the group had engaged in useful dialogue with the technician at an earlier stage during the design process. This would have allowed the group to eliminate various concepts which ultimately proved not to be feasible and were time consuming to investigate. Similarly thinking more carefully about the final assembly of the aircraft at an earlier stage would have allowed a more elegant and lighter solution to be conceived. If the design were to be repeated with these points and the knowledge gained over the course of the project in mind then it is felt that significant weight savings could be made.

Nonetheless, whilst the most ambitious performance target was not met the basic requirements for the aircraft have been successfully fulfilled. It is capable of carrying the maximum allowable 4kg payload within the wings as originally envisaged. It will take-off within the prescribed distance of 61m, have an operating window which is large enough for it to be flown comfortably and will be both stable and controllable at all speeds within this window. Though the final aircraft mass could have been reduced had the design process progressed more smoothly it can be concluded that the concept chosen allowed a light, functional and innovative design to be developed which should perform well at competition.

9.0 References

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