GEOSURV II UNMANNED AERIAL VEHICLE

FINAL REPORT

PART D

STRUCTURES AND MECHANICAL SYSTEMS

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ABSTRACT

The primary objective for the 2008-2009 GeoSurv II Structures group (STR) was to complete the prototype airframe in preparation for first flight. Many aspects of the design, analysis, manufacturing, testing and assembly were completed while adhering to the requirements set forth by SGL and Carleton University. This report presents the work accomplished by the structures group while completing the GeoSurv II prototype airframe.

The fuselage subgroup completed the design and manufacturing of all the fuselage access panels. A method for installing the avionics instrumentation was also implemented. Additionally, the structural interfaces from the fuselage to the engine and from the fuselage to the wing were designed, manufactured and tested.

The wing subgroup completed the manufacturing of both wings for the GeoSurv II prototype. Interfaces, such as the boom attachment, were designed, manufactured, assembled and tested.

The empennage subgroup completed the empennage manufactured during the 2007-2008 academic year. This work included finishing the vertical stabilizers and designing, manufacturing and testing the tail booms.

The flight controls subgroup completed the design and manufacturing of the prototype flaperons. However, the required maximum deflections were not achieved and design changes are necessary. Additionally, the elevators and rudders were designed, manufactured and tested. Testing of the empennage hinge design was performed on an “empennage mock-up”.

The landing gear subgroup completed the design, manufacturing and testing of the nose landing gear. The main landing gear was also manufactured; however, it was determined that the main landing gear could not support the aircraft loads. Consequently, the main landing gear has been redesigned.

The analysis subgroup completed the analysis of key design aspects, including the rear bulkhead of the fuselage. Additionally, a rigging procedure for final assembly was developed, including a colour scheme for painting the aircraft.

The majority of the objectives for the Structures group were completed this academic year. Following the completion of all interfaces, the prototype airframe will be assembled and ground testing will be performed in preparation for first flight.

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1. INTRODUCTION

The objective of the Structures group this academic year was to complete the GeoSurv II prototype airframe (as shown in Figure D-1), in preparation for flight testing. The main structure of the aircraft, with the exception of the nose landing gear, was manufactured out of carbon fiber/epoxy; which was done to reduce the magnetic signature of the aircraft as required by our industry partner, Sander Geophysics Ltd. (SGL). There exist two types of carbon fibre/epoxy: carbon/epoxy monolithic and carbon fibre/epoxy/foam core sandwich. Both types of components were manufactured using either Vacuum Assisted Resin Transfer Moulding (VARTM) or wet lay-up techniques.

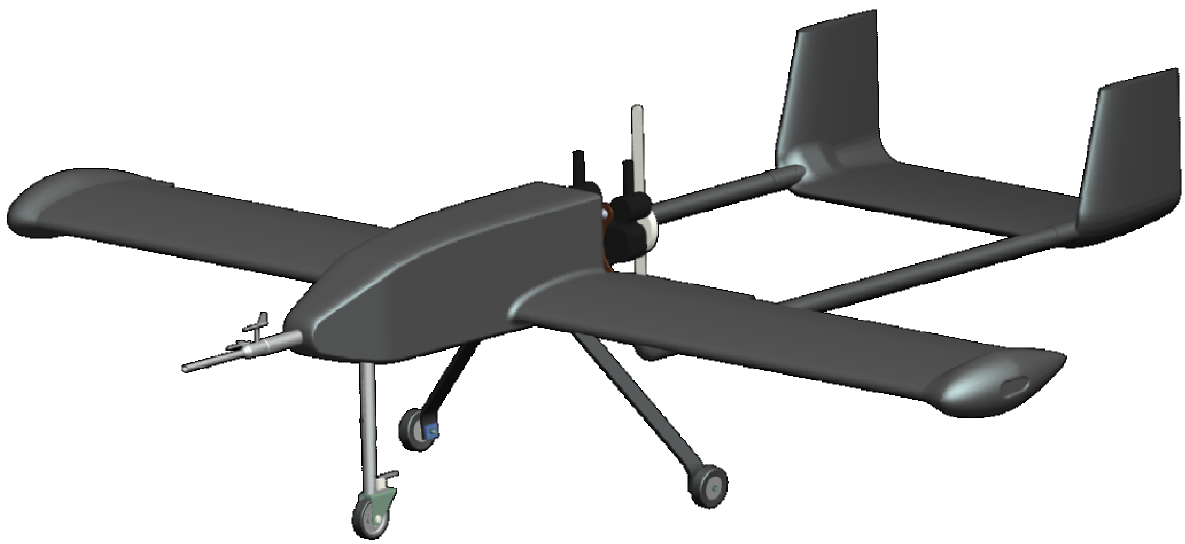

Figure D-1: GeoSurvII 3D Model

To meet the overall objectives, the structures group was divided into six subgroups; fuselage, wing, empennage, flight controls, landing gear and analysis. Due to a time constraints, members of the empennage subgroup were tasked to aid both the wing and flight controls subgroups in the second term. As well, a member of the analysis group was tasked to aid the landing gear subgroup.

The fuselage subgroup was responsible for completing design and fabrication of all access panels, as well as interfacing with the main landing gear (MLG), nose landing gear (NLG), wings, avionics and engine.

The wing subgroup was responsible for completing the manufacturing and assembly of the prototype wings. The wing subgroup was also responsible for interfacing with the fuselage, empennage and flaperons.

The empennage subgroup was responsible for completing the empennage manufactured in the previous academic year, as well as interfacing with the wing, rudders and elevators.

The flight controls subgroup was responsible for completing the design, manufacture and installation of the flaperons, rudders and elevators and associated hardware.

The landing gear subgroup was responsible for completing the design and fabrication of both MLG and NLG and their corresponding interfaces with the fuselage.

The analysis subgroup was responsible for carrying out both Finite Element Analysis (FEM) and analytical calculations for critical areas of the aircraft that experience high loading. The analysis group also was responsible for the rigging procedure of the aircraft as well as the painting procedure.

During the design and fabrication process, each subgroup ensured compliance of their area of responsibility with certification standards. Structural tests were carried out to validate the analysis results, including testing of the wing, tail booms, and nose landing gear.

2. FUSELAGE DESIGN, FABRICATION, ASSEMBLY & TESTING

The fuselage frame, as seen in Figure D-2, was designed and manufactured in the previous academic year (2007-2008) and consists of a foam core with carbon fibre reinforced epoxy skins. This fuselage structure was the basis for all the design work, including interfaces, completed this year. The total weight of the fuselage frame is 11.2 lb with a length of approximately 42 inches. The width of the fuselage is approximately 12 inches at the front and 15 inches at the rear.

Figure D-2: Fuselage Structure from Previous Academic Year

The primary goal of the fuselage group this year was to complete the fuselage structure for the GeoSurv II prototype. This year, all of the access panels were designed and manufactured, excluding the nose cone which was completed last year. A method for installing the avionics instrumentation was designed and manufactured. Furthermore, interfaces for the engine, wings and landing gears were designed, manufactured and installed.

2.1 Access Panels

The access panel requirements for the GeoSurv II prototype were to maintain the outer mould line, keep as lightweight as possible, minimize number of fasteners and ensure quick and independent attachment/removal. As seen in Figure D-3, there are 4 main access panels, as well as the nose cone and the nose cone bridge. The subsequent sections will detail these components.

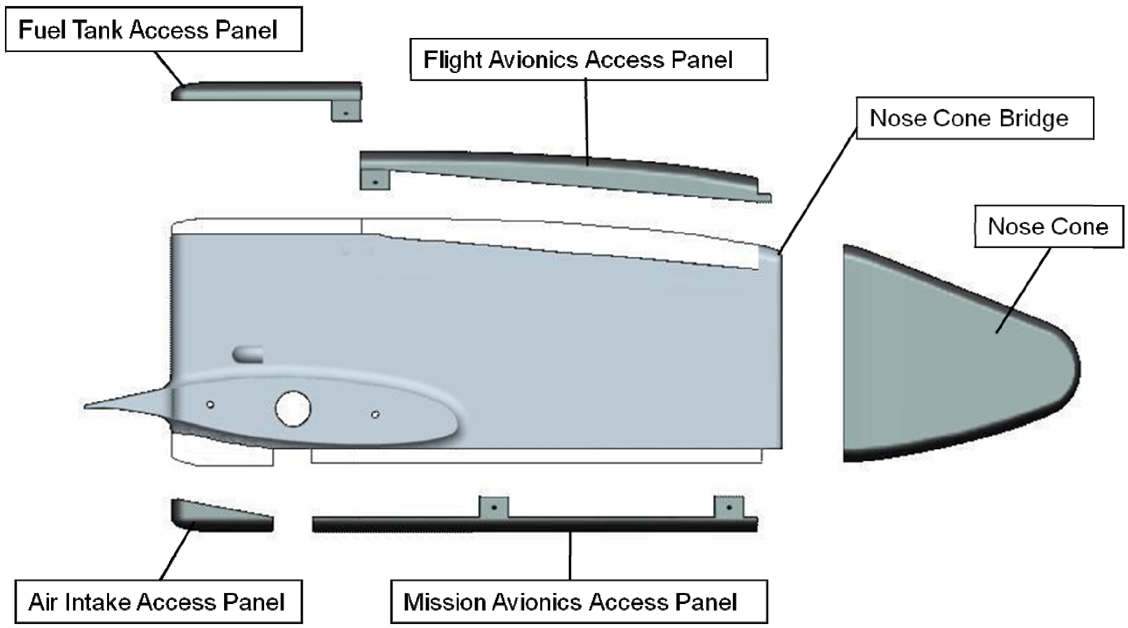

Figure D-3: Access Panel Layout

2.1.1 Nose Cone

The nose cone, which is a carbon fiber shell, was designed and manufactured in the previous academic year [DR 97-02]. It weighs 2.02 lb and sits flush over the nose cone bridge and the sides of the fuselage. The nose cone, as seen in Figure D-4, is fastened twice through the nose cone bridge and once on either side of the fuselage.

Figure D-4: Nose Cone

2.1.2 Nose Cone Bridge

The nose cone bridge, as seen in Figure D-5, is secondary bonded to the front of the fuselage and acts as the interfacing component between the nose cone, fuselage and flight avionics access panel [DR 97-02]. The bridge is the width of the fuselage and consists of a foam core with a carbon fiber skin.

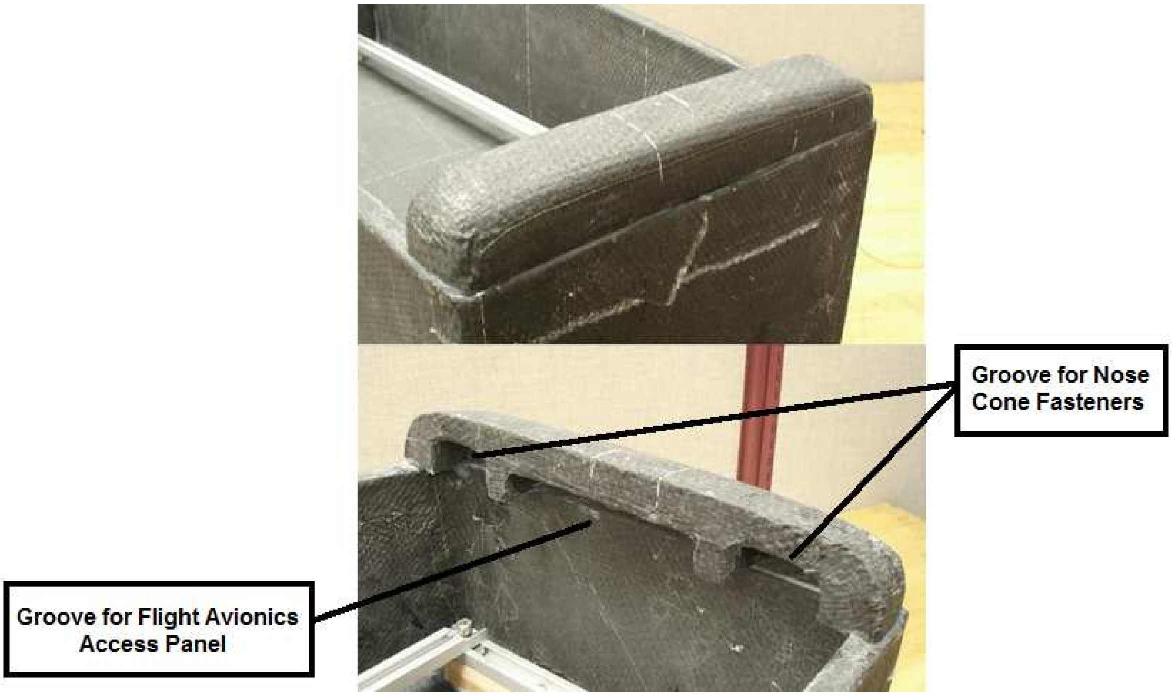

Figure D-5: Nose Cone Bridge

2.1.3 Flight Avionics Access Panel

Located at the front of the upper fuselage, the flight avionics access panel covers the bay containing the flight avionics instrumentation [DR 97-02]. The access panel is a carbon fiber monolithic component with a front extension and a middle support, both of which are a foam core with carbon fiber skin. Due to the front support, this access panel only requires 2 fasteners at the rear, which pass through the brackets on either side. The access panel, as seen in Figure D-6, runs a length of 30 inches and weighs 1.11 lb.

Figure D-6: Flight Avionics Access Panel

2.1.4 Fuel Bay Access Panel

Located at the rear of the upper fuselage, the fuel bay access panel covers the bay containing the fuel tank [DR 97-02]. The final design for the GeoSurv II will incorporate a parafoil in this bay for landing; however, the prototype has not been designed with this feature. The access panel is a carbon fiber monolithic component with 2 brackets at the front through which the panel is fastened. The access panel is retained at the rear with a hinge. As seen in Figure D-7, the access panel is 15 inches long, 15 inches wide, and the total weight of the panel is 0.45 lb.

Figure D-7: Fuel Bay Access Panel

2.1.5 Mission Avionics Access Panel

Located at the front of the lower fuselage, the mission avionics access panel holds the SGL instrumentation [DR 107-02]. This panel has been designed to carry 20 lb, and as such was designed with a foam core and carbon fibre reinforced epoxy skin. The access panel contains 4 brackets through which the panel is fastened. The total weight of the access panel is 1.96 lb and the panel runs a length of 32 inches, as seen in Figure D-8.

Figure D-8: Mission Avionics Access Panel

2.1.6 Air Intake Access Panel

Located at the rear of the lower fuselage, the air intake access panel seen in Figure D-9 covers the bay that provides the air for the engine [DR 107-02]. The engine, with its air intake protruding into the fuselage, draws its required airflow from the air that is routed through the air intake pipes on the sides of the fuselage. Refer to section 2.3.2 for an explanation of this arrangement. The access panel has a groove at the front which allows it to mate with a lip on the main landing gear attachment bracket. A fastener passes through these mated surfaces to retain the panel at the front. A hinge retains the panel at the rear. The access panel, which is only 7 inches in length, has a foam core and carbon fibre skin that weighs 0.68 lb.

Figure D-9: Air Intake Access Panel

2.1.7 Fasteners

The fasteners employed for the GeoSurv II prototype are “Dzus” ¼ turn fasteners [DR 97-05]. The holes in the fuselage have been reinforced with delrin bushings, also used to customize the fastener attachment. As seen in Figure D-10, the fasteners lie flush with the sides of the fuselage thereby maintaining the outer mould line of the aircraft. Additionally, the fasteners are retained within the fuselage walls.

Figure D-10: "Dzus" ¼ Turn Fastener

2.2 Engine Mounting Plate to Fuselage Interface Analysis

An initial concern for the fuselage rear bulkhead was whether the carbon fibre/epoxy/foam core sandwich structure could support the loads produced by the engine. Consequently, an analysis was performed on the initial design of the rear bulkhead to determine if reinforcement of the fuselage was necessary. This analysis led to the final design of the rear bulkhead as presented in sections 2.3 and 2.3.1.

An analysis was performed on the rear bulkhead design. The purpose of the analysis is to determine if additional reinforcement is required at the engine mount standoffs and the engine air intake hole, due to stress concentrations and material deformation.

2.2.1 Model Construction

The Finite Element (FE) model was constructed with a combination of shells and solids. The carbon fibre skin was modelled as shell elements, while the foam core, engine mount plate, and standoffs were modelled as solid elements. See DR 97-04 for detailed model assembly. Figure D-11 shows the major components of the FE model.

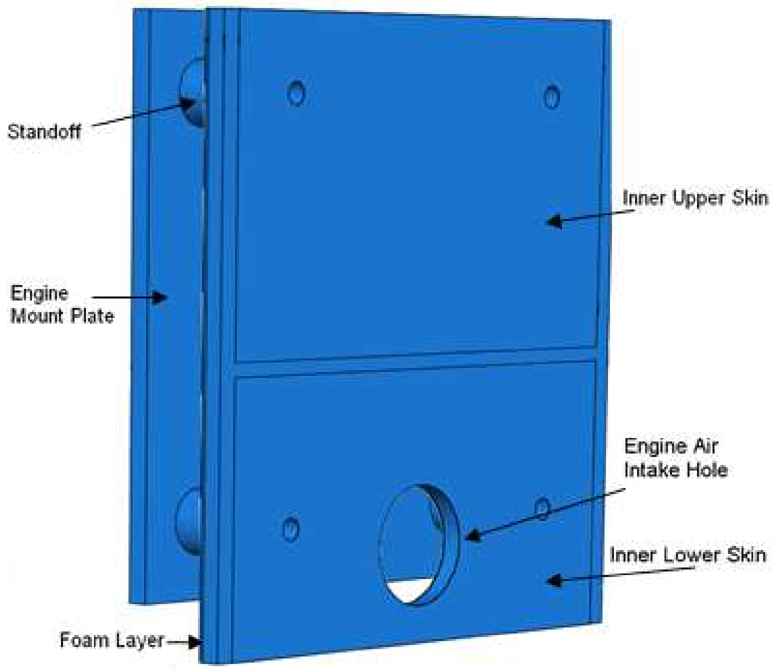

Figure D-11: Rear Bulkhead FE model

The material properties for the carbon fibre/epoxy and the foam core were obtained from DR 87-05. See DR 97-04 for stainless steel properties.

The FE model consists of only the rear bulkhead of the fuselage, the engine mount plate, and the engine standoffs. Only the rear bulkhead was only considered from the fuselage in order to obtain a finer mesh. If the entire fuselage was modelled, the elements would be too large to obtain an accurate result because ABAQUS limits the number of elements that can be used in the mesh. Smaller meshes show more accurate results on the modelled components. To act as the fuselage supporting structure, the rear bulkhead is simply supported along the cross section. The results of the analysis are conservative, because the stresses and deflections will be larger than in reality.

2.2.2 Loads

The rear bulkhead must withstand the thrust loads, from the engine propeller, and the weight and inertial loads of the engine. A safety factor of 6.75, obtained from DR 83-02, was applied to the forces. The forces were converted into pressures and applied to the engine mount plate. Table D-1 describes the magnitude, direction, and the area where the force acts. Figure D-12 summarizes the applied loads.

Table D-1: Ultimate Loads

Loads	Force (lb)	Applied Pressure with safety factor (psi)	Applied Location
Thrust	200	64	1 inch Radius Circle in middle of Engine Mount Plate, Forward
Weight	239	1428	Bottom Edge of Engine Mount Plate, Downwards

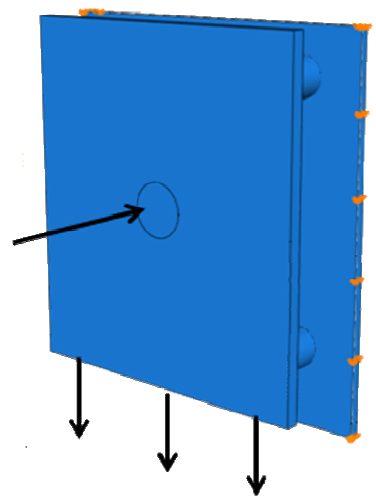

Figure D-12: Ultimate Loads

2.2.3 Results

2.2.3.1 Tensile Stress

The maximum tensile stress was found to be 90 ksi, located around the bottom standoffs. This is shown in Figure D-13. The stress is increased at the bottom standoffs due to the engine air intake hole. The removal of the material weakens the surrounding structural strength. This maximum tensile stress, however, is below the ultimate tensile strength of the carbon fibre/epoxy. Therefore, the rear bulkhead is able to withstand the forces in terms of stress.

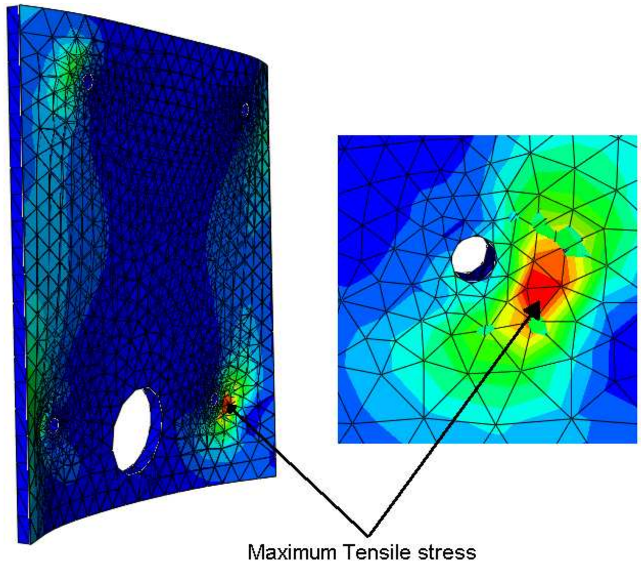

Figure D-13: Maximum Tensile Stress

2.2.3.2 Maximum Deflection

The maximum deflection occurred around the engine air intake hole on the rear bulkhead. The surface of the rear bulkhead deflects 0.28 inches at the intake hole inwards towards the fuselage. The top of the rear bulkhead deflects 0.14 inches outwards of the fuselage. The difference in deflection is caused by the engine air intake hole. The removal of the material weakens the composite structure. The deflection at the engine air intake hole is too large, and with the combination of the engine vibration can lead to structural failure. The rear bulkhead must be reinforced around the engine air intake hole. The modified rear bulkhead is described in Section 2. The deflection can be seen in Figure D-14.

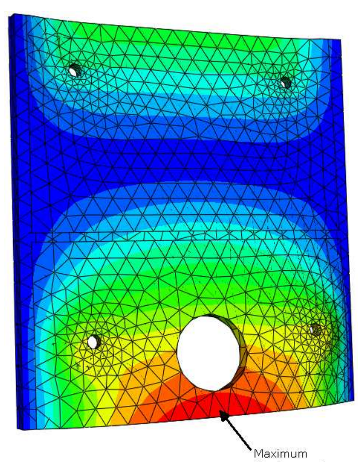

Figure D-14: Maximum Deflection

2.3 Overview of Interface for Prototype Following Analysis

For the GeoSurv II Prototype, the engine mounting plate is attached to the fuselage as seen in Figure D-15 and Figure D-16. The fuselage subgroup was responsible for the design, manufacturing and assembly of the standoffs highlighted in Figure D-16.

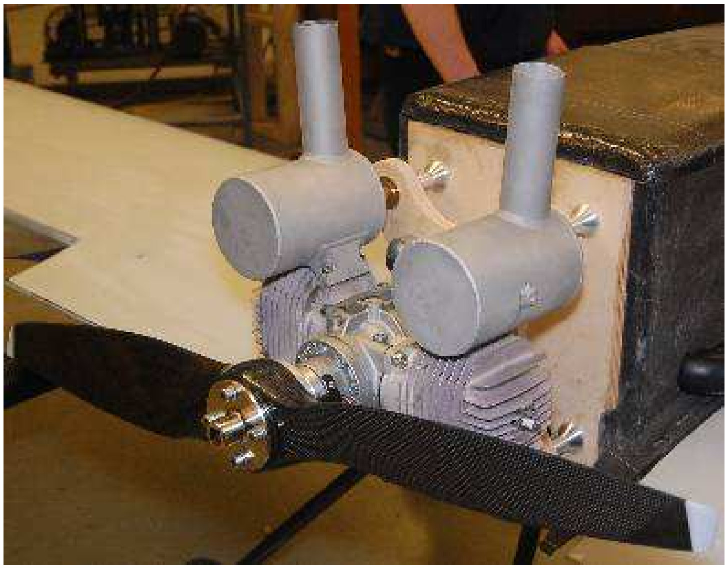

Figure D-15: Engine to Fuselage Attachment

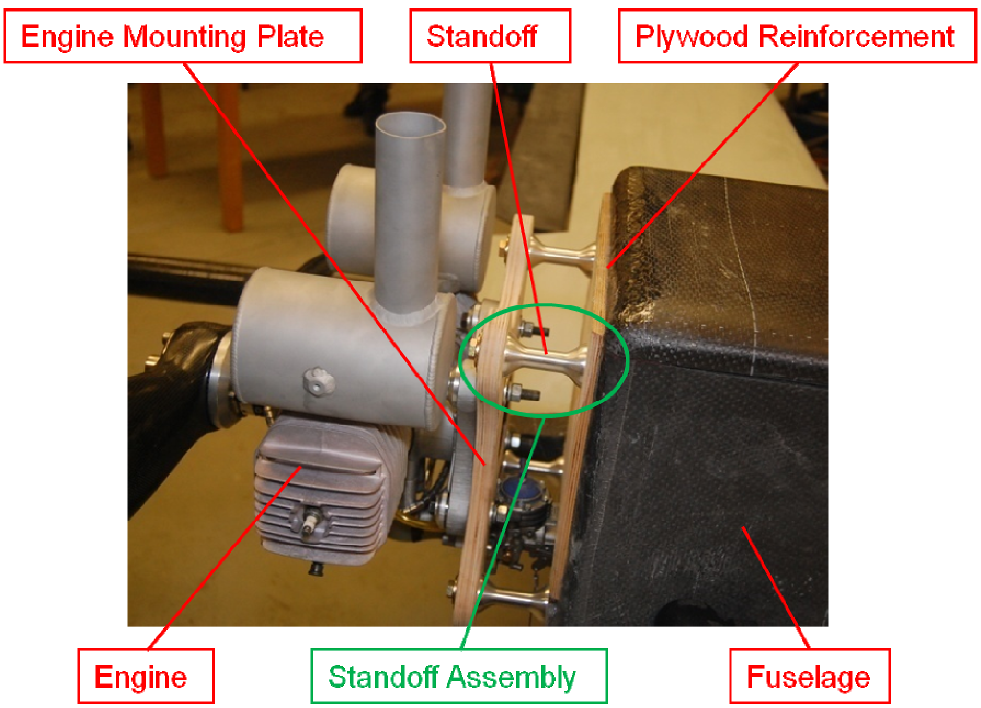

Figure D-16: Overview of Engine Mounting Plate to Fuselage Interface

2.3.1 Design of Standoff Assembly

A cross section view of the standoff assembly can be seen in Figure D-17 [DR 107-01]. The subsequent sections will provide an overview of each of the standoff assembly components.

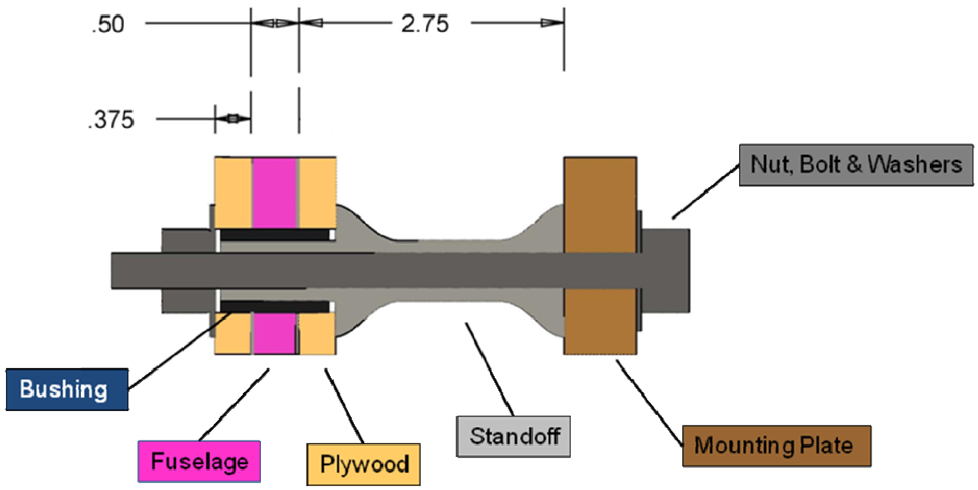

Figure D-17: Standoff Assembly (inches)

2.3.1.1 Plywood Reinforcement

The fuselage structure has been reinforced on the inside and outside with aerospace grade plywood to support the structure during dynamic loading [DR 107-01]. Due to the unknown performance characteristics of the carbon fibre/epoxy under dynamic loads imposed by the engine, this conservative approach was deemed necessary to ensure safe operation of the prototype. It was also a recommendation from the rear bulkhead analysis presented in Section 2.2. The plywood was secondary bonded to the fuselage using Room Temperature Vulcanisation silicone adhesive. The plywood prior to installation on the fuselage weighed 2.592 lb.

2.3.1.2 Bushing

The standoff assembly bushing was manufactured out of delrin and secondary bonded into the sandwich structure [DR 107-01]. The bushing covers the foam core of the fuselage and provides a method of load transfer between the standoff and the fuselage/plywood sandwich structure. Each delrin bushing weighs 0.012 lb.

2.3.1.3 Standoff

The mounting plate is separated from the fuselage by a custom standoff component [DR 107-01]. The standoffs were manufactured out of aluminum 6061, and they have a light interference fit with the delrin bushings. Each standoff component weighs 0.132 lb.

2.3.1.4 Nut, Bolt & Washers

Four AN bolts connect the engine mounting plate to the fuselage/plywood sandwich structure [DR 107-01]. The bolts are 3/8 inch diameter, weigh 0.174 lb and have a light interference fit with the standoffs. The bolts are fastened with aerospace grade self locking nuts and a 1.375 inch diameter washer on both the bolt head and nut.

2.3.2 Air Intake

To maintain an appropriate center of gravity, the engine needed to be as close to the rear bulkhead of the fuselage as possible. Unfortunately, the location of the air filter and carburetor prevented this from happening. As a result, a section of the rear bulkhead was cut away and the air filter and carburetor extend within the fuselage. This cut away also provides sufficient clearance for the movement of the carburetor during engine vibration [DR 107-01]. The cut away can be seen in Figure D-18.

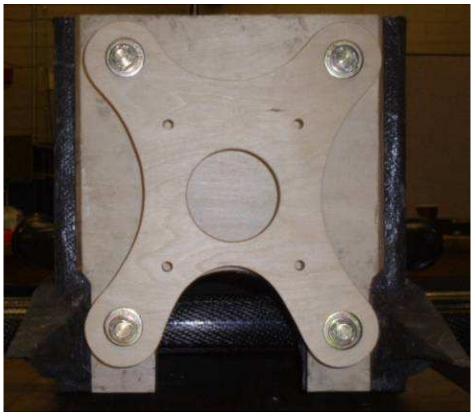

Figure D-18: Cut Away in Fuselage for Engine

With the air intake for the engine positioned inside the rear of the fuselage it was necessary to route external air into this compartment [DR 107-01]. A plastic elbow was secondary bonded into the fuselage, thereby allowing air to flow into the air intake compartment for the engine. The cut away is large enough to ensure there is not a build up of air in the compartment. The weight of each air intake is 0.12 lb and the installed intakes can be seen in Figure D-19.

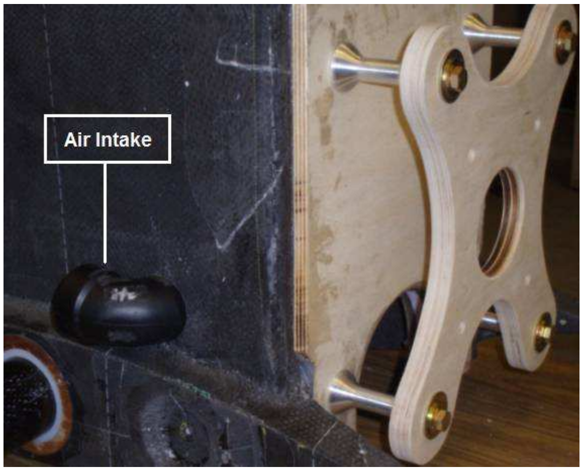

Figure D-19: Air Intake for Engine

2.3.3 Testing

The weight and thrust of the engine imposes both static and dynamic loads on the fuselage structure. An overview of both static load testing and dynamic load testing is discussed in the subsequent sections.

2.3.3.1 Static Load Testing

A test section of the fuselage was manufactured and one standoff assembly was installed and tested [DR 107-01]. The test section, with its foam core, carbon fibre skin and plywood reinforcement, can be seen in Figure D-20.

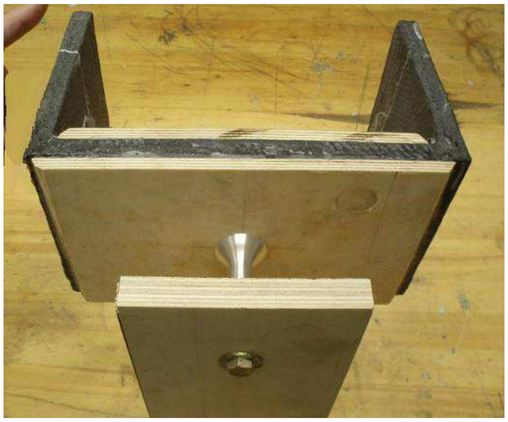

Figure D-20: Test Section for Static Loading

For a single standoff assembly, the test section was required to carry 50 lb. This included the 20 lb weight of the engine with the design ultimate load factor of 10.125. The test reached 400 lb without failure, at which point the test was stopped. Permanent deformation of the plywood, as seen in Figure D-21, was experienced at 200 lb, well past the 50 lb requirement.

Figure D-21: Plywood Reinforcement Deformation from Deflecting Standoff

As a result, the structure was deemed more than sufficient for carrying the static loads imposed by the engine.

2.3.3.2 Dynamic Load Testing

Testing the fuselage structure for the dynamic loads imposed by the engine was not performed because there was no configuration of a test setup that would yield representative results. Since the engine vibration will ultimately be transmitted to the fuselage and the aircraft as a whole, testing one small section would be unrepresentative.

The plywood reinforcement was installed to allow this vibration without damaging the aircraft structure. The plywood was also recommended by the engine manufacturer as the preferred method of attachment. With the experience of the engine manufacturer, and the fact that similar sized model aircraft employ a similar method of reinforcement, the current design is expected to support the dynamic loads imposed by the engine [DR 107-01].

2.4 Fuselage to Wing Interface

The purpose of this interface is to transfer lift and manoeuvring loads generated by both the wing and empennage. The following sections detail the design and implementation of the wing interface.

2.4.1 Loads

The wing interface was designed to carry 3.25×10^4 in-lb of bending moment, shear loads of 1000 lb and a pitching moment of 8056 in-lb [DR 107-07]. The Wing interface was also designed to allow removal of the wing for ground transportation.

To carry the bending moments and shear loads, a tubular carry through spar has been used. The carry through spar is 48 in long with an inner diameter and outer diameter of 2.225 in and 2.625 in respectively [DR 107-11]. The carry through spar is shown in Figure D-22.

To carry the pitching moments, two custom shear bolts have been used. The bolts are positioned 13 in apart and one bolt carries 720 lb while the other carries 624 lb [DR 107-07].

Using both the carry through spar and the shear bolts enables the aircraft to be modular, as shown in Figure D-22.

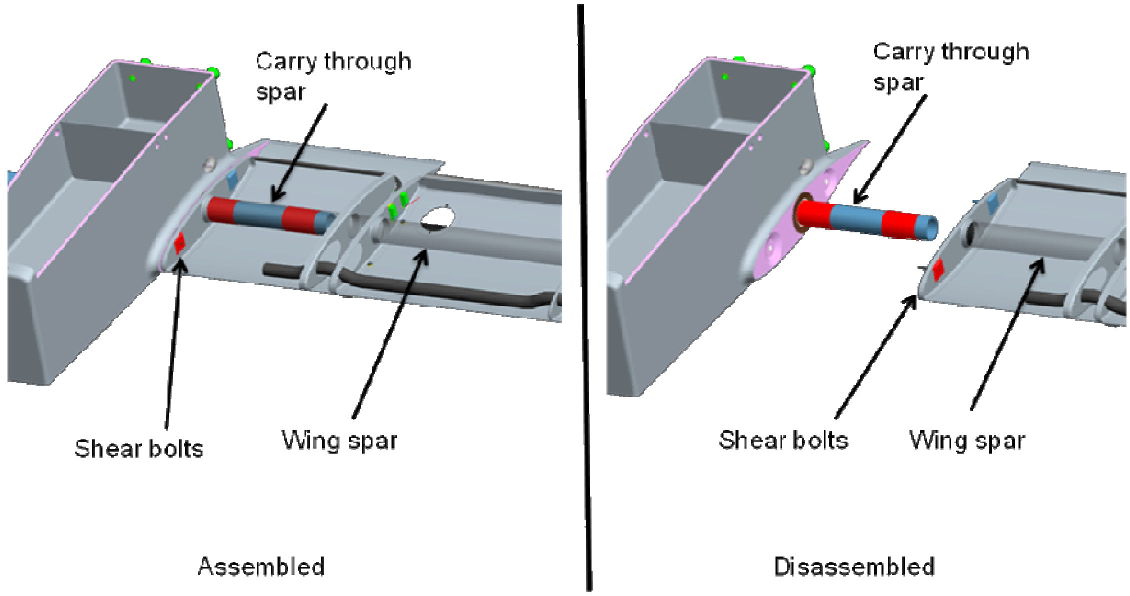

Figure D-22: Wing Interface

2.4.2 Carry Through Spar Analysis

The purpose of the carry through spar is to determine the minimum number of plies required for flight. The composite structure is a woven carbon fibre/epoxy material with a [0°/90°] ply lay-up.

2.4.2.1 Model Construction

The FE model was constructed using shell elements and partitioned edges. A detailed model assembly can be shown in DR 97-18.

Assembly Parts

Figure D-23 shows the FE model of the carry through spar.

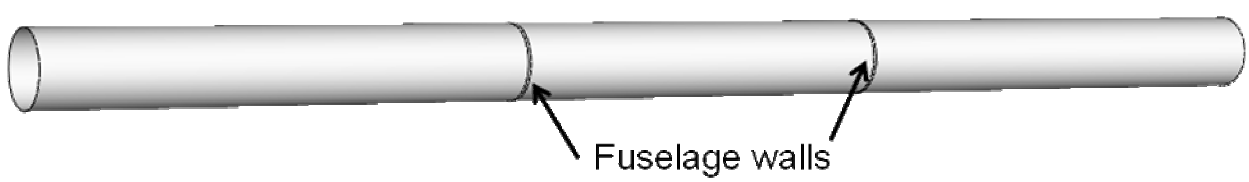

Figure D-23: Carry Through Spar FE model

Material Properties

The material properties for the carbon fibre/epoxy and the foam core were obtained from DR 87-05.

Boundary Conditions

The FE model consists of only the carry through spar. The fuselage is represented by partitioned edges along the carry through spar. Since the purpose of the analysis is to determine the minimum number of plies required, the effect the spar had on the fuselage was not considered. To simplify the analysis, the carry through spar experienced no deflection within the fuselage. The resulting stresses will be greater in the analysis than in reality. The carry through spar is simply supported where the fuselage side walls exist.

Loads

The carry through spar must be able to withstand ultimate design loads. The applied loads are the maximum bending moment and the maximum shear force. The loads were obtained from DR 77-08. A detailed load analysis is shown in DR 97-18. The maximum force applied to the tip of the carry through spar is 1911.76 lb.

2.4.2.2 Results

The analysis was performed for several iterations in order to determine the minimum number of plies required to withstand ultimate design loads. The stress analysis resulted in a requirement of 8 plies. The maximum stress induced in the carry through spar, 112.5 ksi, was located just outside of the fuselage walls. The stress shown at the tips of the carry through spar are secondary forces that exist due to the application of the loads onto the spar. The maximum tip deflection is 3.3 in. Figure D-24 displays the maximum stress in the 8-ply spar, and Figure D-25 displays the maximum tip displacement.

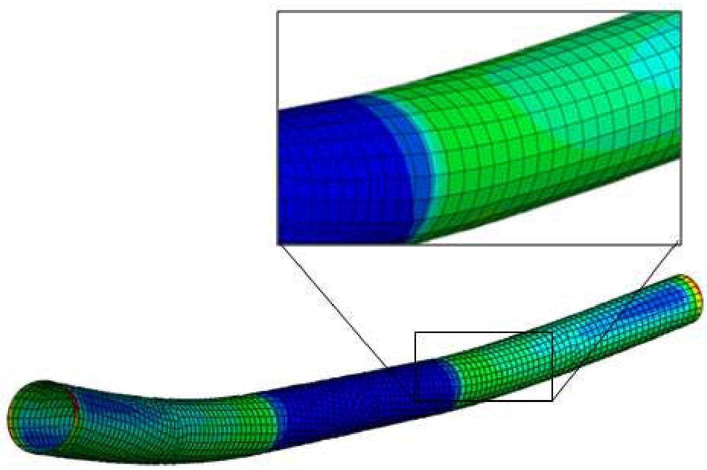

Figure D-24: Maximum Stress

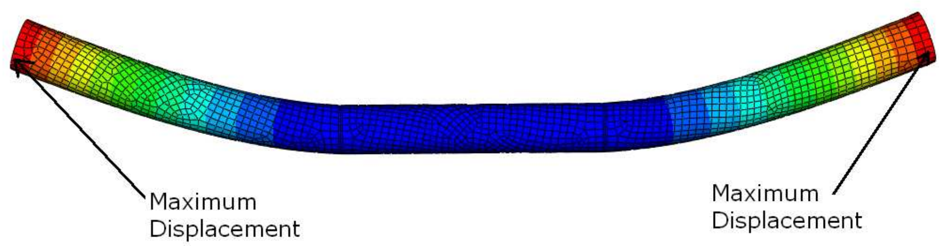

Figure D-25: Maximum Displacement

2.4.3 Interface Design

The carry through spar was designed to be removable, resulting in two bushings on either side connecting it to the fuselage fairings [DR 107-11]. The two bushings were manufactured out of wood and delrin. The carry through spar is also equipped with two outer diameter bushings. These bushings allow for a light interference fit between the carry through spar and the wing spar, preventing relative motion between the spars.

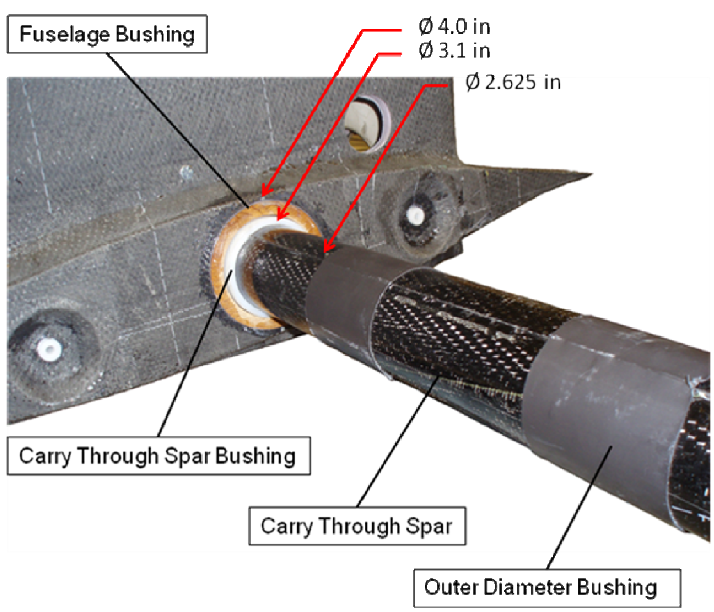

Figure D-26: As Built Wing Interface

The shear bolts have also been designed to be removable from the wing. This was done to allow replacement in the event of damage, see section 3.2.2 for more details. The design of the custom bolts required a modification to the fuselage. The modification was a creation of a pocket within the fuselage fairing as shown in Figure D-27. The depth of the pocket is 0.5 in and has a taper to the edges of the fairing, reducing stress concentration points [DR 107-10]. To support the shear bolt, a delrin bushing was added at the centre of the pocket, as shown in Figure D-27. The bushing is equipped with a flange on one end to prevent the pull out of the bushing in the event that it catches thread during removal of the shear bolt.

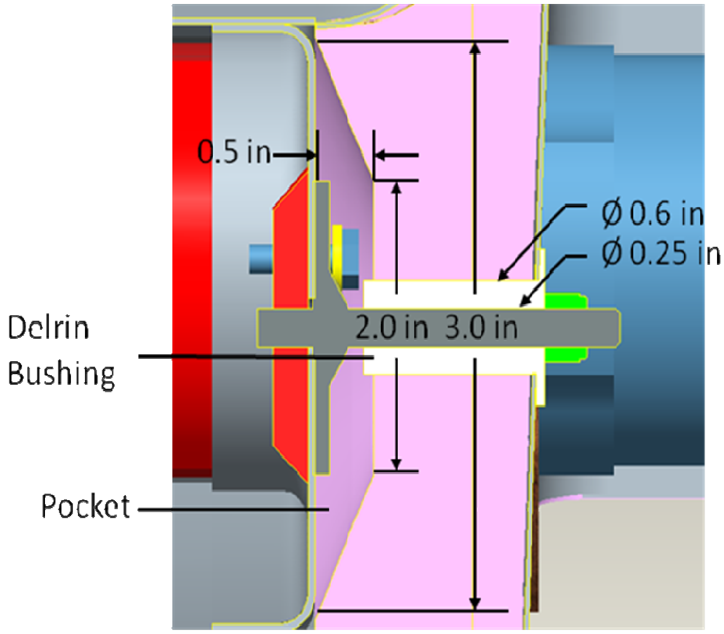

Figure D-27: Fairing Cross-section

2.4.4 As Built Configuration of Fuselage to Wing Interface

To meet the requirements imposed by the design of the shear bolt, a 4 in hole was drilled for the carry through spar and two 0.6 in holes were drilled for the shear bolts.

At the location of the shear bolts, pockets were carved out, as described in section 2.5.2. The pockets were locally reinforced using 4 plies of carbon fibre/epoxy with varying diameters and infused directly within the structure, as shown in Figure D-28. The complete interface after infusion and bonding of all bushings is shown in Figure D-29.

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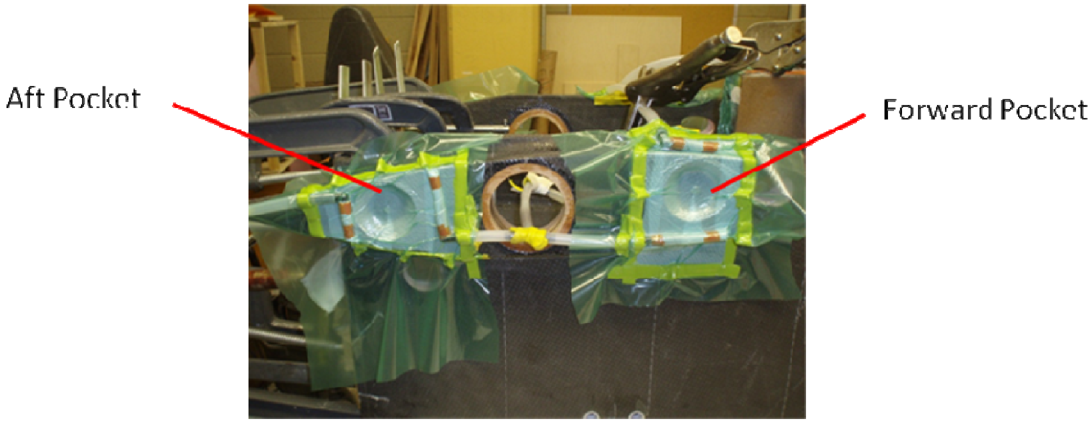


Figure D-28: Fairing Pocket Infusion

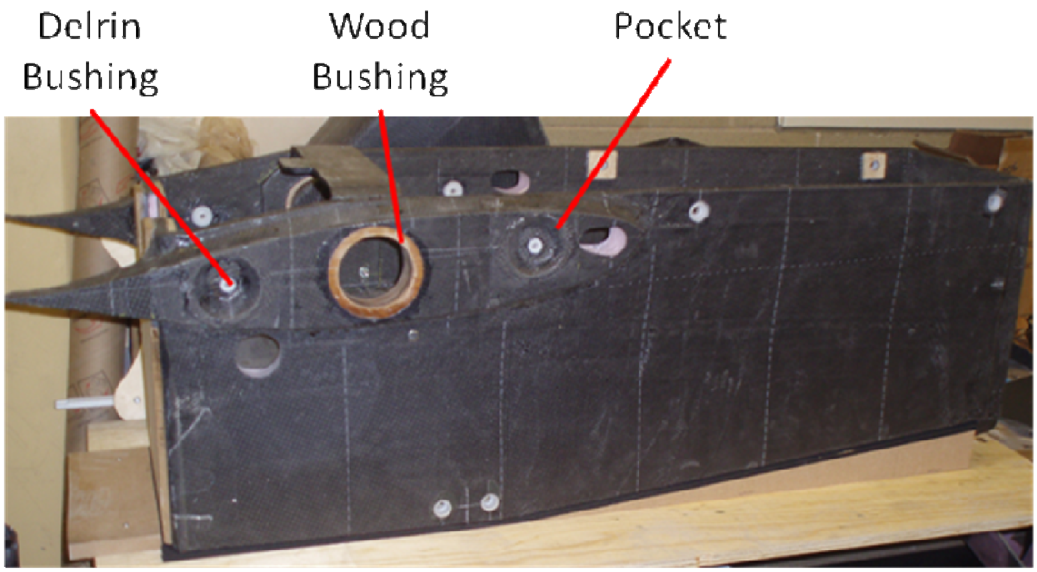


Figure D-29: As Built Fuselage Wing Interface

2.5 Fuselage to Avionics Interface

Throughout this academic year, a design for mounting the avionics into the fuselage was developed and installed. The driving requirement for the design was to have a versatile mounting system allowing for multiple avionic configurations with a single fuselage. The subsequent sections describe the design and installation of this interface.

2.5.1 Interface Design

To install multiple avionics boxes and mounting boards within the fuselage, a rack was designed to fit inside both the flight and mission avionics bays. Avionics boxes attach to the rack, rather than the fuselage H-Structure, therefore eliminating the need for holes within the H-Structure.

The rack was designed to withstand a crash load of 21g [DR 107-08] and designed to utilize standard components [DR 107-08]. Components used for the design can be seen in Figure D-30 and are described in DR 107-08. All but the custom wood base are standard off-the-shelf components.

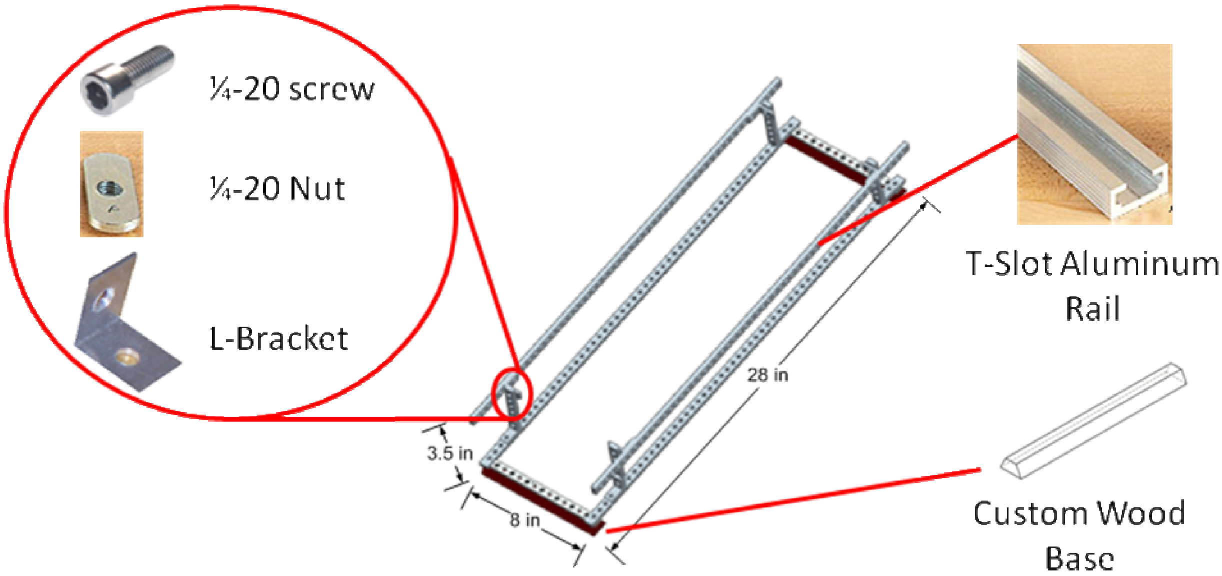


Figure D-30: Rack Sub-Components

2.5.1.1 Flight Avionics Rack Configuration

For the flight avionics rack, the components were assembled as outlined in DR 107-08 to form the assembly shown in Figure D-31(A). The rack was then attached to the custom wood blocks which were bonded to the fuselage structure, as seen in Figure D-31(B).

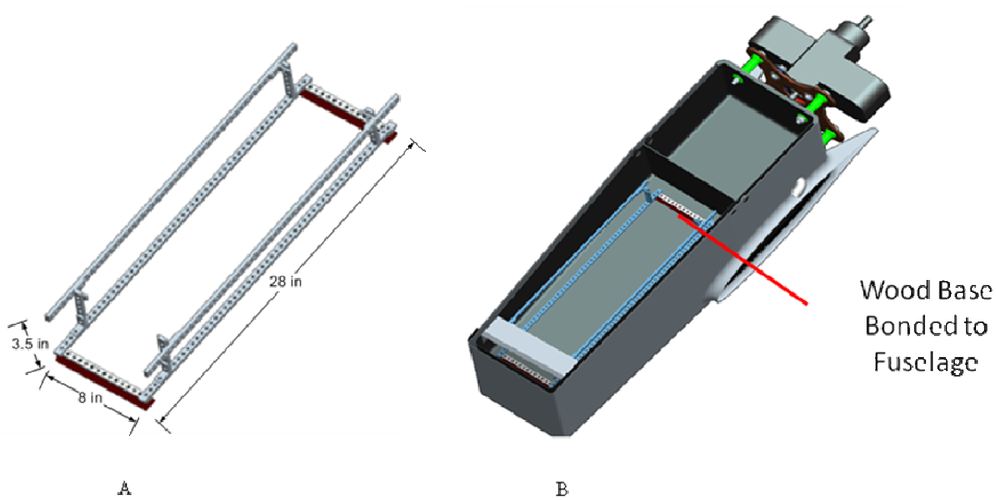


Figure D-31: Assembled Flight Avionics Rack

2.5.1.2 Mission Avionics Rack Configuration

For the mission avionics rack, the components were assembled as outlined in DR 107-08 to form the assembly shown in Figure D-32 (A). The rack attaches to the custom wood blocks which are bonded to the access panel as shown in Figure D-32(B).

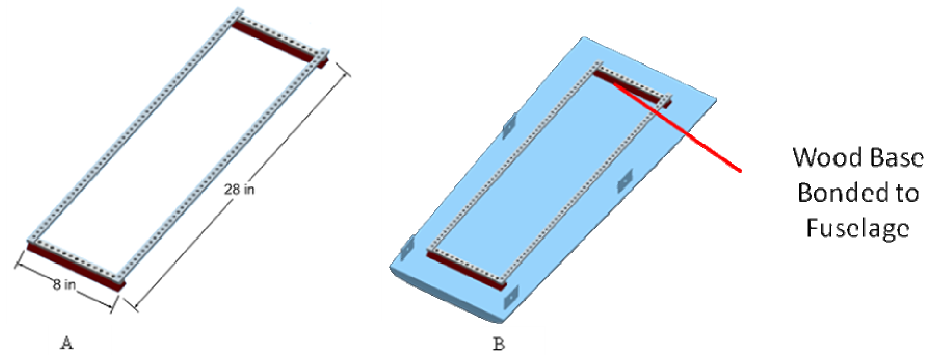


Figure D-32: Assembled Mission Avionics Rack

2.5.2 As Built Avionics Interface

Manufacturing of the flight avionics bay rack was completed this year; however, the rack was installed with a single level (shown in Figure D-33)since there was no immediate need for a second level by the avionics group.

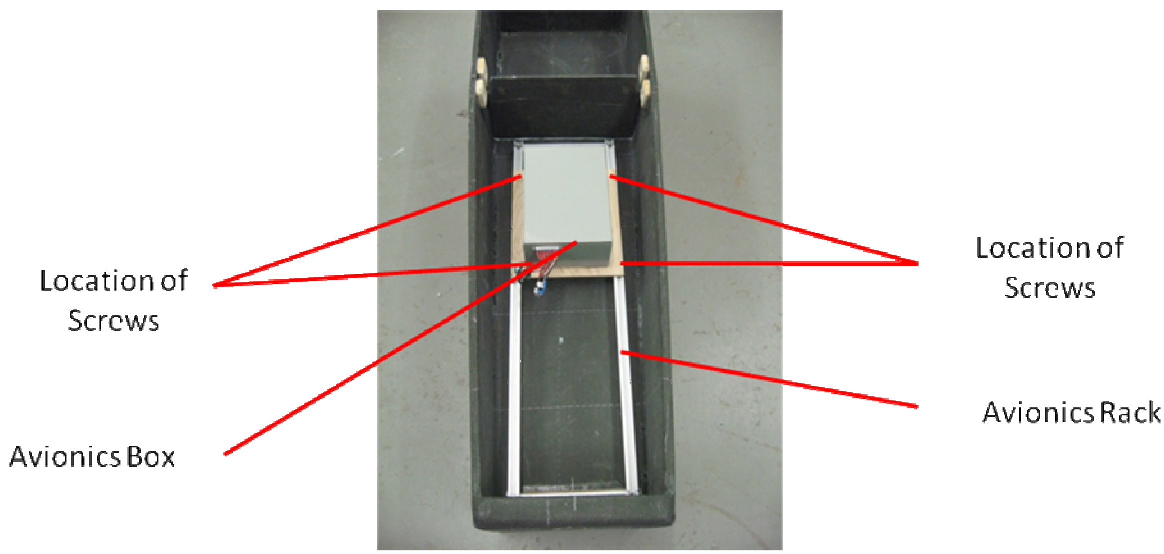


Figure D-33: As Built Avionics Installation

3. WING DESIGN, FABRICATION, ASSEMBLY & TESTING

This academic year, significant progress was accomplished on the GeoSurv II prototype wings. The detail designs of the interfaces were completed, several key elements were manufactured, assembly was successfully accomplished, and important structural testing was performed. The following section will describe the design, fabrication, assembly, and testing of the wings.

3.1 Overview

Each GeoSurv II prototype wing is approximately 7.4 ft long, 2.2 ft wide and weighs 19 lb. The main structural elements include a tubular spar, six ribs and an upper and lower skin. Each wing interfaces with the fuselage and empennage, and contains a flaperon and magnetometer. All of these elements are shown in Figure D-34.

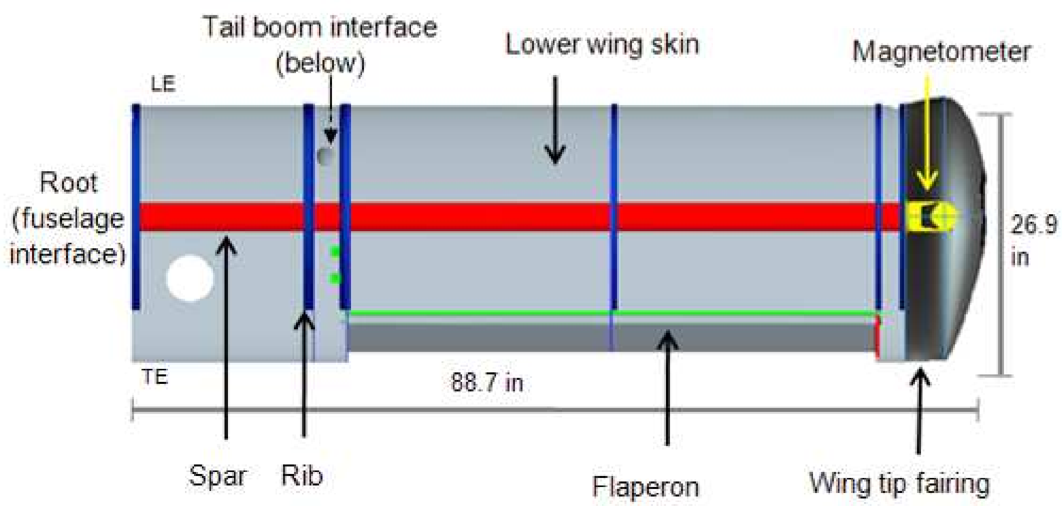


Figure D-34: GeoSurvII prototype wing top view

3.2 Detail Design

The following section provides an overview of the wing-boom interface, wing-fuselage interface, shear bolts and the avionics layout.

3.2.1 Wing- Boom Interface

The empennage boom is contained in a cradle comprised of two carbon fibre components namely the outer and inner cradles (Figure D-35). Both fasten to the lower wing skin. The outer cradle forms a tight fit around the sides and the underside of the boom. This outer cradle also contains two shear bolts that go through the middle of the boom and prevent them from pulling out during flight. The inner cradle forms around the top of the boom and sits between the outer cradle and the wing skin. The two cradles are sandwiched together and then fastened using AN4 bolts and lock washers to the underside of the wings (Figure D-35) so they are in line with the empennage-boom interface pockets.

Both the cradles have two flanges that contain a total of six bolt holes. The cradle flanges line up with the flanges of two wing ribs. The bolt holes are matched in the wing skin and two wing ribs. A threaded titanium nutplate is bonded to the inside of the wing rib using a secondary bonding adhesive (Figure D-36). This design allows the cradles to be removable. Also, it is a FAR 23 requirement that no secondary bonding is used along primary load paths and that each fastener uses two retaining devices. The full details of the interface can be found in DR97-16.

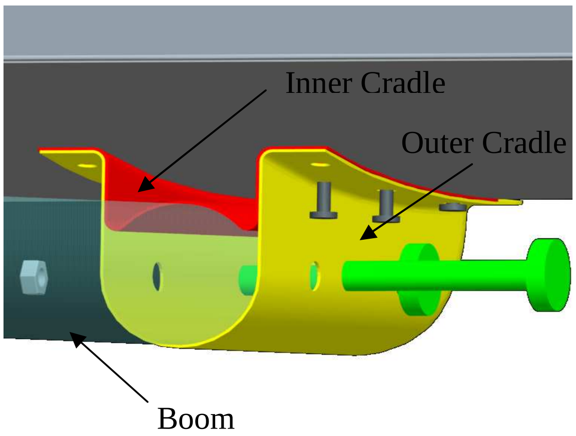


Figure D-35: Wing-Empennage Interface

Figure D-36:Interface bolt and nutplate

3.2.2 Wing-Fuselage Interface Shear Bolt Design

The wing to fuselage interface is responsible for joining the wing to the fuselage. It also transfers loads from the wing and empennage into the fuselage. The interface consists of two major components, one being the carry through spar discussed in Section 2.5 and second being the shear bolts. This component of the interface consists of a custom shear bolt that is fastened to the wing rib. The cross section of this part of the interface is shown in more detail in Figure D-37.

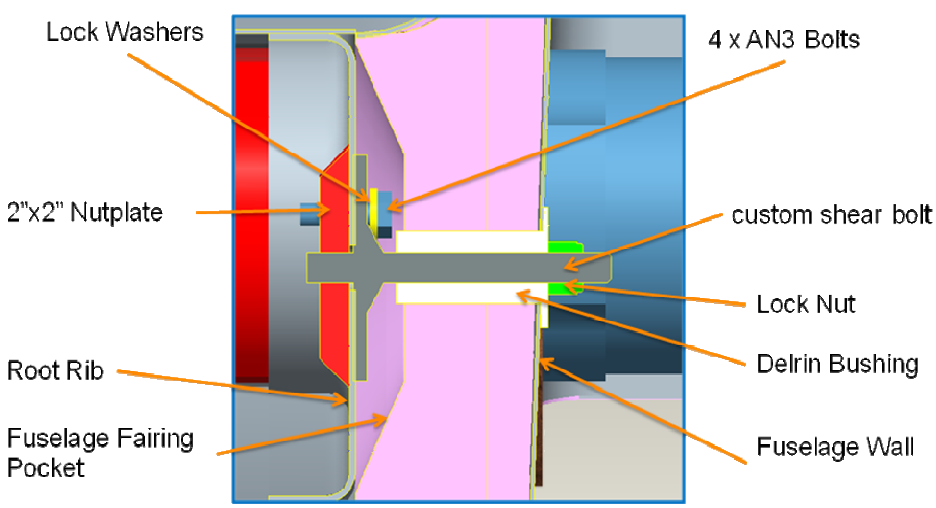


Figure D-37: Wing-fuselage interface cross section

The custom shear bolts (Figure D-38) are fastened to the root rib at two locations in order to carry the shear loads generated by the empennage and wing twist. This bolt slides through the fuselage wall and is held in place on the inside of the fuselage using a lock nut.

The custom shear bolt consists of a 0.25in shaft with a 2in diameter plate welded on to it. The plate contains four smaller radial, clearance fit holes. The short end of the bolt slides through a tight fitting hole in the root rib and then into a nutplate. Four AN3 bolts are added through the radial holes of the plate to ensure that the main bolt is retained and sits flush against the rib.

Figure D-38: Wing-fuselage interface custom bolt

The nutplate (Figure D-39) is bonded to the inside of the root rib. Holes in the nutplate match the holes in the bolt's plate with the radial holes being threaded. The carbon fibre is put over top of the nutplate to ensure that the nutplate will stay retained if the adhesive fails.

Figure D-39: Wing-fuselage interface custom nutplate

The interface conforms to FAR23-607 and FAR23-611 which state that fasteners must use two retaining devices and that the bolts have to be accessible so they can be inspected and maintained. The full details of the interface are given in DR107-07.

3.2.3 Avionics Interface

The layout of wing avionics for the GeoSurv II prototype was established as displayed in Figure D-40. The magnetometers will be mounted at the wing tips, and the sensor electronics (pre-amplifiers) will be placed in the fuselage lower mission avionics bay for convenient accessibility and simplicity. A fixed length cable connecting these components will detach at the pre-amplifier end, and will be protected in the wing by a 1.25 in diameter plastic conduit that runs along the leading edge. The empennage wiring will pass through the same conduit and travel through a small access hole in the lower skin located above the boom cradle. The two flaperon servos, which are mounted on the third rib from the root of each wing, are accessible via an access hole in the lower skin. Servo wiring is contained in a ½ in diameter conduit that runs along the leading edge and curves in order to reach the servos. Details and selection of this configuration are provided in DR97-13.

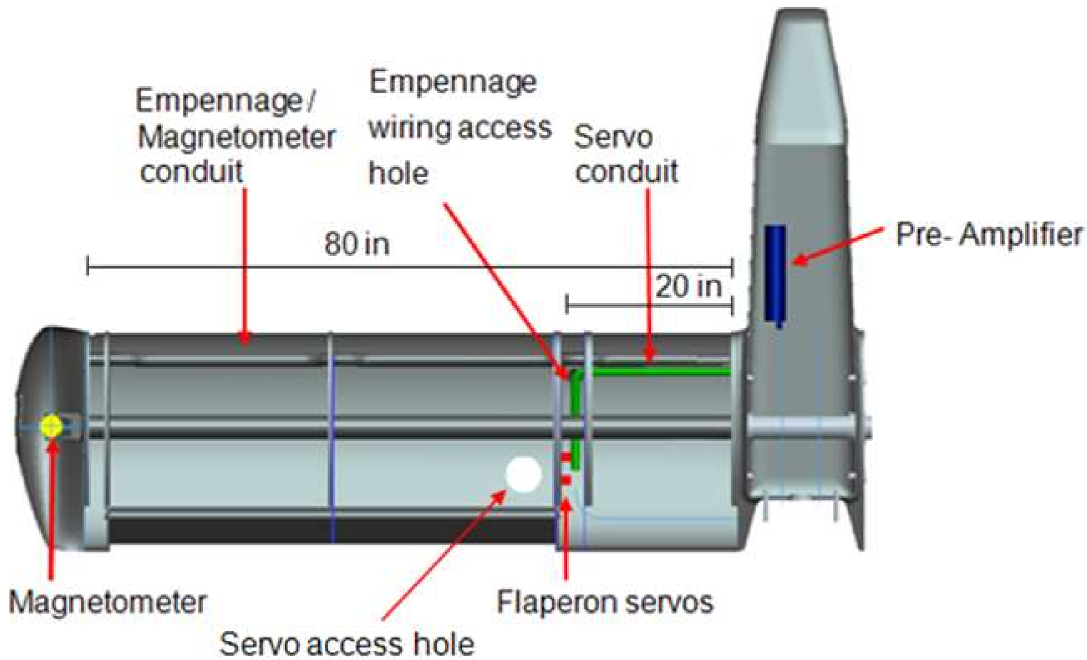


Figure D-40: Wing-avionic configuration

3.3 Wing Component Fabrication

3.3.1 Lower Wing Skins

The lower wing skin is the second element of a two part design forming the complete prototype wing skin. Two high-quality lower wing skins were successfully fabricated and are shown in Figure D-41.

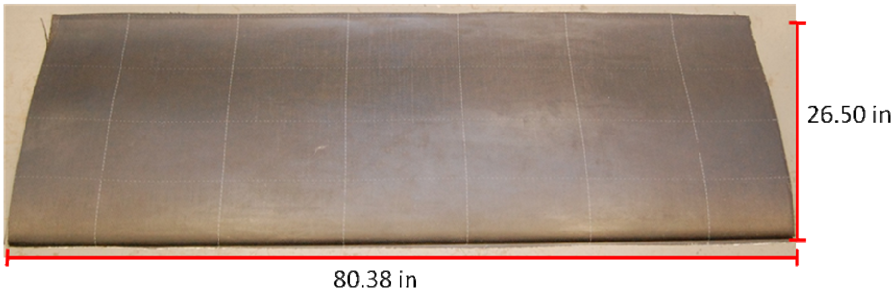


Figure D-41: Lower wing skin

To fabricate the wing skins, a full scale lower skin mould was machined from medium density fibreboard. The mould fabrication process was defined using computer aided manufacturing, and then machined by a computer numerical control (CNC) machine as described in DR97-01.

Once the mould was completed , the lower wing skins were fabricated using VARTM. The setup included laying up two plies of carbon fibre on the mould (in the 0/90 and +/-45 fibre orientation), followed by a layer of peel ply, distribution medium, and finally vacuum bag as described in DR97-01. The part was then infused by applying vacuum pressure to the outlet (labelled B) which pulled mixed epoxy from the inlet (labelled A) through the part as displayed in Figure D-42. The final step was then to allow the part to cure at room temperature for 12 hours. Further details about the fabrication process as well as problems encountered and recommendations are described in DR97-01.

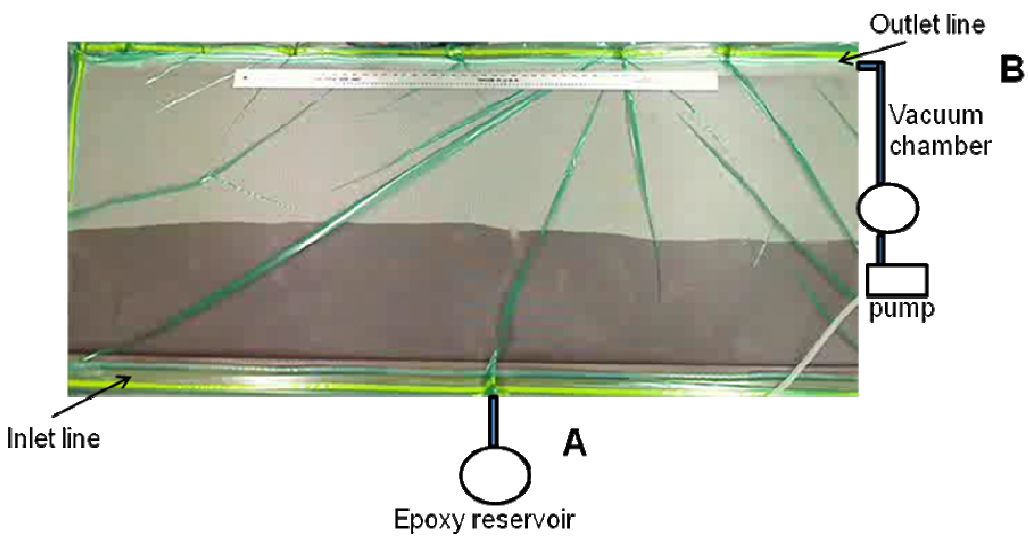


Figure D-42: Top view of lower wing skin infusion setup and process

3.3.2 Wing Tip Fairings

The wing fairings are located at the tip of each wing and serve to protect, host, and provide accessibility to the magnetometers. The fairing comprises of two components, an inboard fairing which is permanently bonded to the tip rib (rib 6) and an outboard fairing which is fastened and removable as shown in Figure D-43.

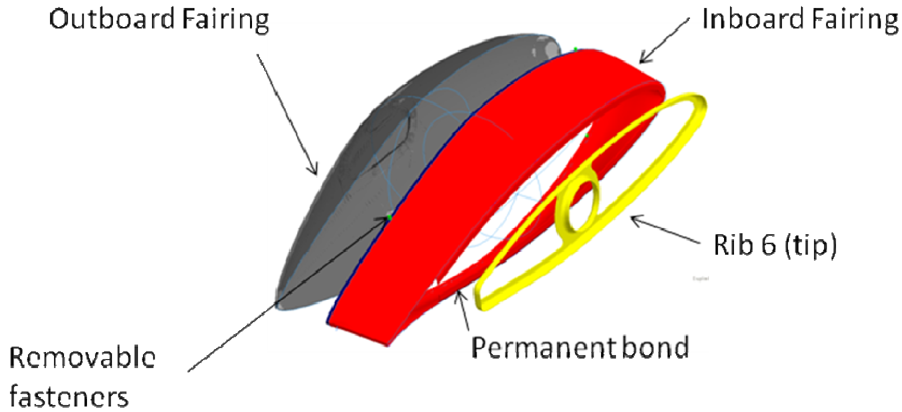


Figure D-43: Exploded view of wing tip fairing elements

For simplicity and efficiency it was attempted to manufacture a fairing as a single part by VARTM from a male foam mould (Figure D-44). However, when the complete part was cut in half and the mould removed, the two components distorted and no longer fit together properly.

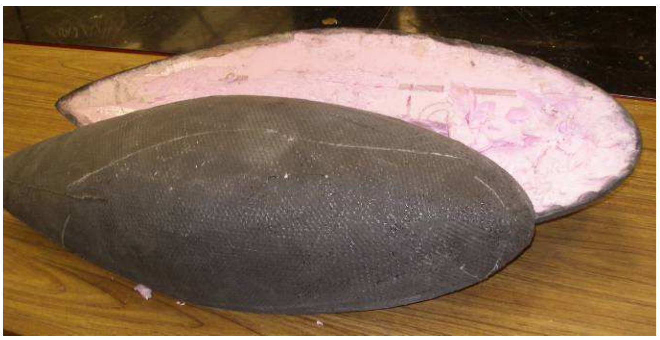


Figure D-44: Manufacturing attempt of inboard and outboard fairing from a male foam mould

It was decided not to progress with the current manufacturing process. Two new fairings will be fabricated, possibly using a female mould, and two components that overlap. The complete design, manufacturing and recommendations on the wing tip fairings are available in DR107-03.

3.4 Wing Assembly

The following process, including cutting and bonding, was executed to successfully assemble both wings.

3.4.1 Assembly Process

The lower wing skin mould served as the primary wing assembly jig, with the datum as the root and leading edge corner as seen in Figure D-45.

3.4.1.1 Lower Wing Skin

The first step was to align and clamp the lower skin to the root and leading edge of the mould (Figure D-45).

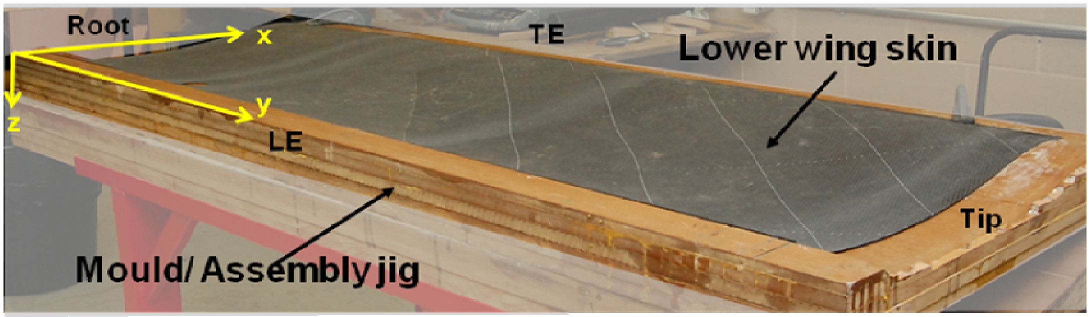


Figure D-45: Lower wing skin

3.4.1.2 Ribs

The six wing ribs were then bonded, as described in 3.4.2 to the lower wing skin. Rib assembly fixtures were accurately placed along the lower wing skin mould and were used to properly position and align each rib (Figure D-46).

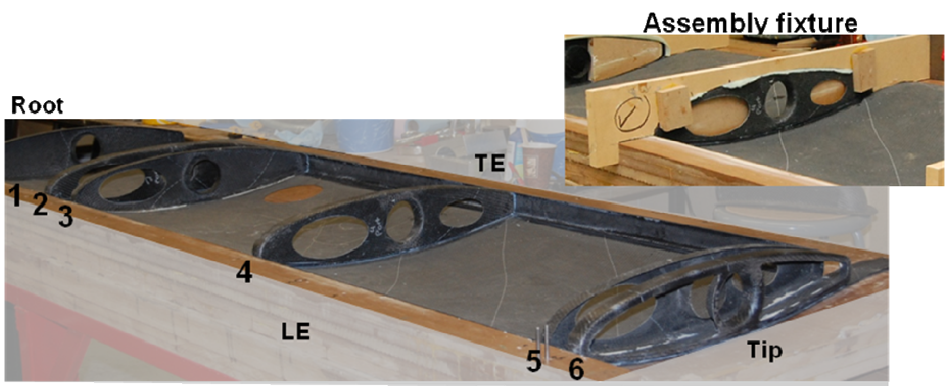


Figure D-46: Wing rib assembly

3.4.1.3 Flaperon Interface

To fill and support the flaperon cut out at the trailing edge, a close out spar running the flaperon length and two triangular trailing edge caps at either end were bonded to the lower assembly (Figure D-47).

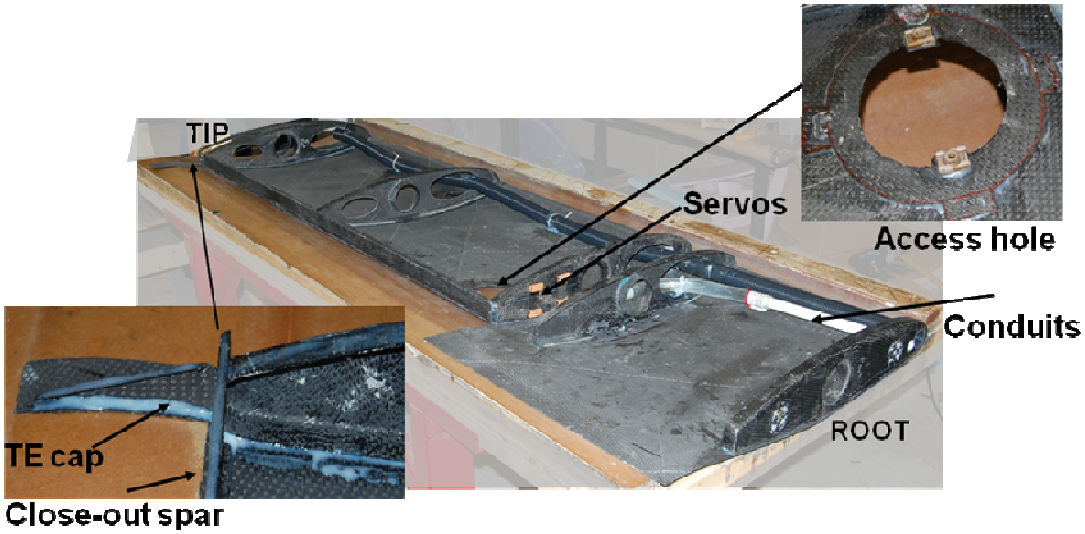


Figure D-47: Wing flaperon and avionic interface assembly

Further, to provide accessibility to the servos mounted on the third rib, a 5 in diameter access hole was cut out of the lower wing skin. A lip was bonded around the hole, to which the cut out was fastened back on to create an access cover.

3.4.1.4 Avionics Interface

The conduits and corresponding configuration described in Section 3.2.3 were installed onto the lower wing assembly. The bonding adhesive did not adhere to the plastic conduits; therefore carbon fibre strips were wrapped around the conduits and bonded to the surrounding ribs and skin.

3.4.1.5 Fuselage Interface

To install the fuselage interface shear bolts a central hole was drilled at the required rib locations. Then the bolt was slid into the hole and the radial holes were drilled. Following this, the nutplate was bonded to the inside of the rib with the shear bolt and four radial bolts in place to ensure alignment. Finally, the nutplate was covered with carbon fibre. The process is shown in Figure D-48.

Figure D-48: Wing-fuselage interface assembly

3.4.1.6 Boom Interface

The cradles were installed on the lower surface of the wing by first locating their chordwise and spanwise locations on the wing. All six flange bolt holes were drilled through all components and then nutplates were bonded to the inside of the wing ribs. Bolts were in place during curing in order to ensure proper alignment. The finished product is shown in Figure D-49.

Figure D-49: Wing-Empennage interface assembly

Once the UAV is assembled 1/8 in pilot holes were drilled through the outer cradles at the location of the shear bolts. After disassembly these holes were enlarged to ½ in diameter and fitted with bushings. A 5/16 in hole was drilled through the bushing to finish the interface.

3.4.1.7 Spar

One of the final elements was the installation of the tubular spar which was permanently bonded to the six ribs (Figure D-50). The bonding adhesive was used at a high viscous state to achieve a uniform bond along the full circumference.   A significant issue that arose with the spar assembly was the creation of unintentional wing twist, as a result of bonding not occurring in the assembly jigs. This issue was addressed in the bonding of the upper skin.

Figure D-50: Wing spar bonded to ribs

3.4.1.8 Upper Skin

The final critical step was the addition of the upper skin. To proceed with the upper skin, all previous steps had to be complete as following this step there is limited access to the wing interior. Significant measures were taken to properly align and position the skin. Each skin was bonded to c-channels along the leading edge, wood inserts at the trailing edge, and the upper rib flanges. Due to the resulting wing twist from the spar, the lower wing skin had to be clamped into its required position and the upper skin was then used to hold this configuration in place. As a result, there are residual shear stress along the ribs, however the magnitude is minimal and spread out along the entire rib flange area. The final product is shown in Figure D-51. Full assembly details including fixtures, process, and problems encountered are available in DR107-15.

Figure D-51: Wing upper skin assembly

3.5 Bonding

3.5.1 Rib to Wing Skin Bonding

The assembly of the wing components (ribs and wing skins) was conducted using Hysol 9430 adhesive. Before this assembly took place, a bonding process was developed by conducting three bonding trials. The purpose of these bonding trials was primarily to verify that a satisfactory bondline thickness could be achieved. In general, as the thickness of a bondline increases, the mechanical properties of the bond decrease. For this reason, a target bondline thickness range of 0.01 – 0.02 in was selected.

3.5.2 Bonding Trials

Three bonding trials were performed in order to verify that satisfactory bonding could be achieved between the ribs and the wing skins. Bondline thickness data was collected from each of these trials and was used to evaluate the bonding process. Bonding trials 1 and 2 highlighted the requirement for bonding fixtures to ensure proper component alignment during bonding. The third bonding trial resulted in a slight modification to the bonding fixture design (see section 3.5.3 for further information on the bonding fixtures). The data collected from the third binding trial is shown in Figure D-52.

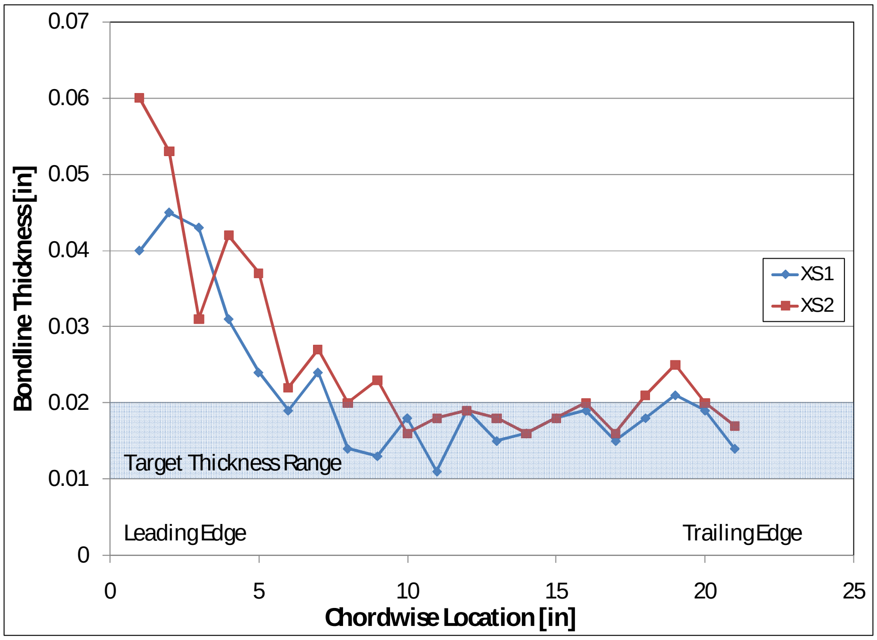


Figure D-52: Bondline thickness data from bonding trial 3

Further information on the bonding trials can be found in DR107-19.

3.5.3 Bonding Fixtures

The final bonding fixtures used in the assembly of the wings consisted of four major components: the wing skin mould, the rib supports, the rib inserts and the wing spar. The bonding fixture set-up is shown in Figure D-53.

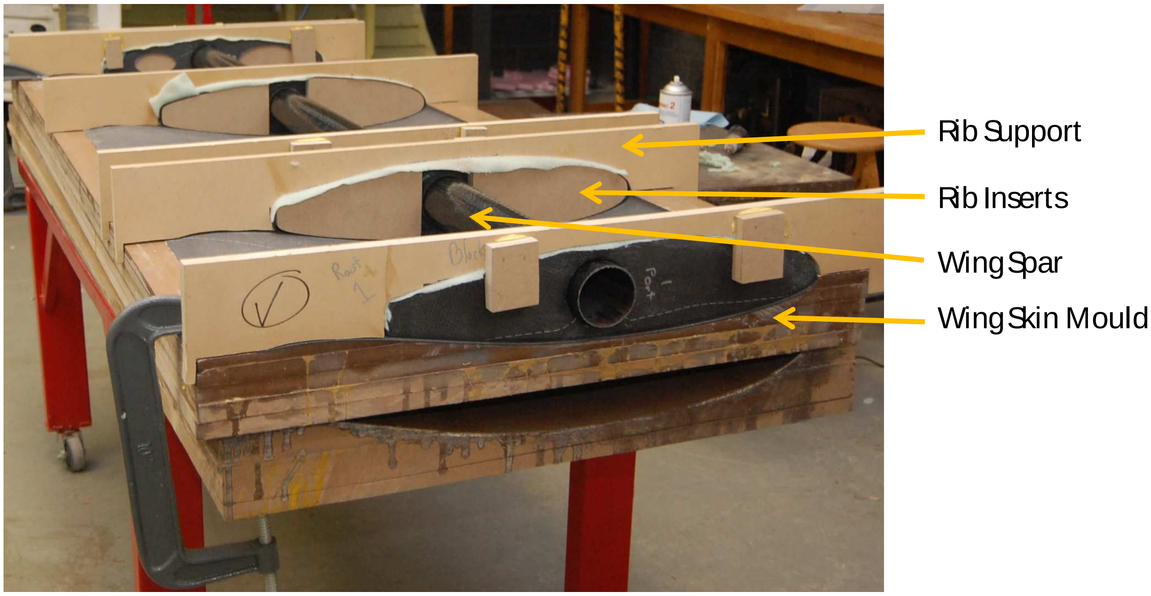


Figure D-53: Bonding fixture set-up

The rib supports were used to apply downwards pressure on the ribs during bonding, and they also ensured the ribs did not slide backwards due to the downwards pressure. The rib inserts were used to ensure the rib flanges did not distort due to the applied pressure during bonding. The wing spar was used to ensure the ribs were properly aligned with respect to each other. Finally, the wing skin mould was used for two reasons: (i) the wing skin sits flush with the mould making it easier to work on, and (ii) the rib supports were indexed along the span of the mould to ensure the spanwise placement of each rib was correct.

Similar fixturing was used for the bonding of the upper wing skin; however slight modifications were made to the rib supports to allow for the upper wing skin to fit within the fixturing. Also, the rib inserts were not used for the bonding of the upper wing skin as it would not be possible to remove them after bonding was complete. This did not result in any adverse distortion of the ribs during bonding.

3.6 Cutting and Trimming of Major Wing Components

Each wing component manufactured using VARTM required trimming in order to achieve its final shape and reduce weight. This was a significant and challenging task of the assembly process. Components that were trimmed include the wing leading edges, wing root and tips, flaperon cut-out, access holes, and the boom cradles. The full details on how each component was trimmed are presented in DR107-24.

3.7 Wing Testing

3.7.1 Nutplate Pull-Through Test

In order to ensure the structural integrity of the wing-boom interface, it was required to determine at what load the nutplate will pull through the carbon fibre/epoxy. This was determined using a test section that replicated the wing skin and the wing ribs found at the interface. A nutplate the same size as the one present in the full interface was installed.

A fastener, screwed into the nutplate, was then attached to a bracket. A weight platform was then hung from this bracket. Weights were added to the platform incrementally until the interface failed at 505lb. At failure the rib buckled in, the adhesive between the rib and the skin failed and the nutplate pulled through the 2-ply rib.

The ultimate load on a bolt is 450lb thus the design is adequate. In order to ensure that the bolt head does not pull through the cradles, flat washers will be added. The setup and test results are shown Figure D-54. The full details of the test can be found in DR107-25.

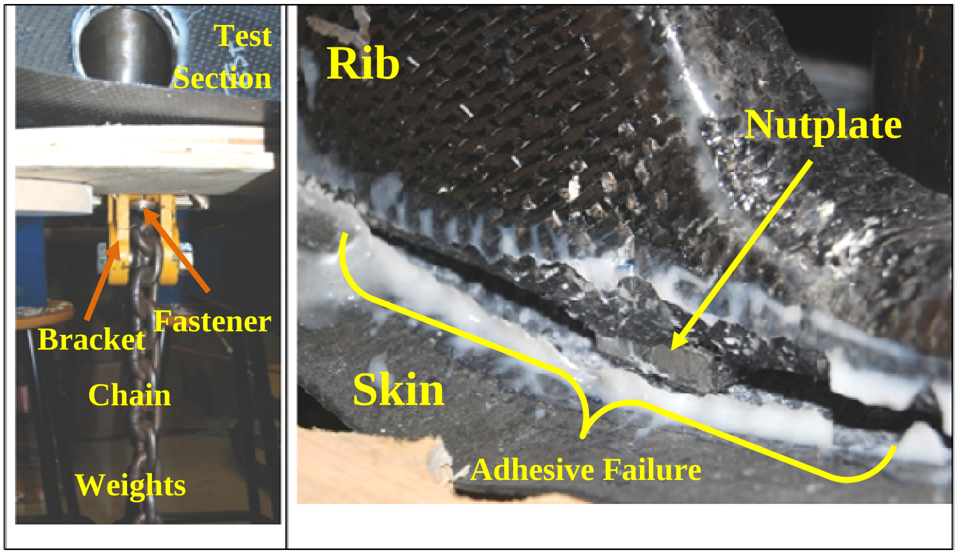


Figure D-54: Wing nutplate pull-through test setup and test results

3.7.2 Bearing Test

A static bearing test was conducted in order to determine the bearing capacity of a composite laminate. This was intended to replicate the loads of the shear bolts on the root rib of the fuselage interface, which sustain a maximum load of 720 lb.

The test consisted of statically loading the edge of a 4-ply composite laminate panel with a close fitting AN-5 bolt, as seen in Figure D-55. Several 5/16 in test holes including carbon fibre and graphite and silica reinforcements were tested.

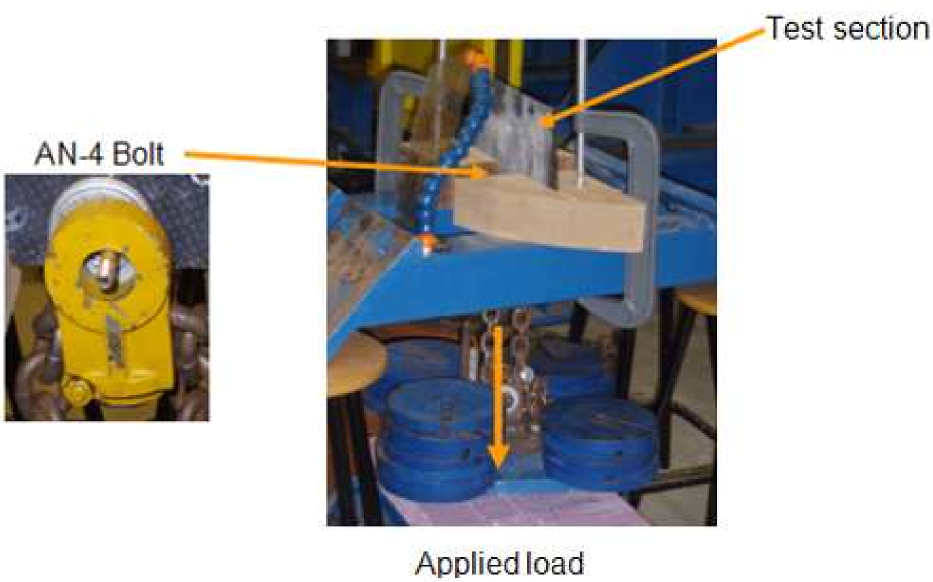


Figure D-55: Wing bearing test

The static bearing strength of a 4-ply composite panel edge loaded with a 5/16 in bolt, whether reinforced or not, is 450lb (Figure D-56). This implies the interface would fail prior to ultimate load and thus the UAV as presently built cannot exceed a load factor of 4.2. This was determined to be satisfactory for initial prototype flights, however is an important limitation.

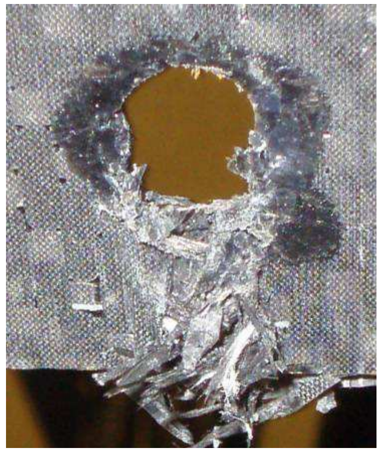


Figure D-56: Failed 4-ply composite panel

Further reinforcement was considered, which would require adding additional plies of carbon fibre to the already manufactured and assembled root rib. However, the quality of the bond and additional infusion would be uncertain and could not be adequately assessed, thus could not be depended on. Full test setup, results and conclusions are available in DR-107-16.

3.7.3 Structural Testing

Initial structural testing was executed on the GeoSurv II prototype wings and fuselage to ensure they can withstand the loads expected during first flights. The testing consisted of applying static loads on both wings up to 50% of the design limit and measuring the corresponding tip deflection. The maximum testing load was determined to be 338 lb per wing which was distributed over the ribs in accordance with the aerodynamic load distribution as seen in Figure D-57.

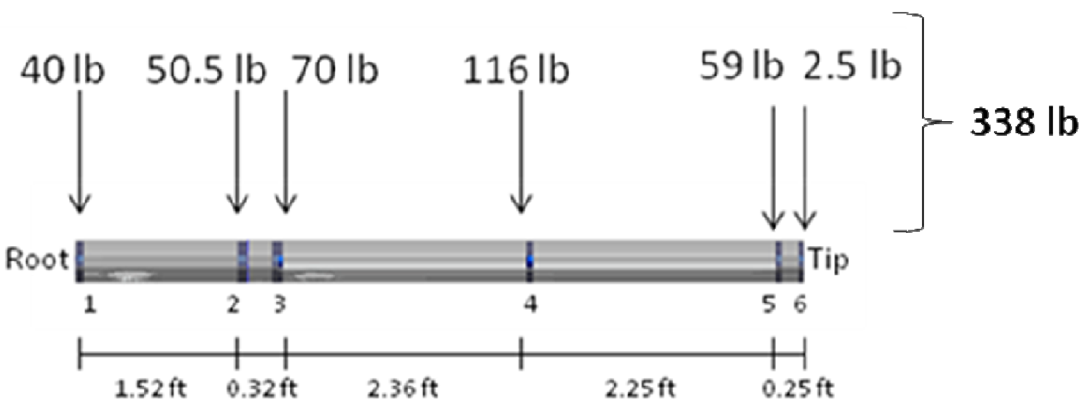


Figure D-57: Maximum test load per rib (50% limit load)

The wing and fuselage were assembled in their flight configuration and inverted. To minimize the loads experience on the fuselage walls, a custom fuselage mount was fabricated which distributed the loads along the entire the entire fuselage length. A sheet of plywood was laid across the lower wing surfaces which were loaded incrementally with sandbags at various loading cycles (Figure D-58). At the maximum load, the resulting tip deflection was determined to be 3.75 in. Full flaperon deflection was possible at all loads. One concern of note was a small clearance that formed along the upper perimeter between the two carry through bushings which is being investigated by the fuselage team. Complete load breakdown, test setup, process, and full results are available in DR-107-17.

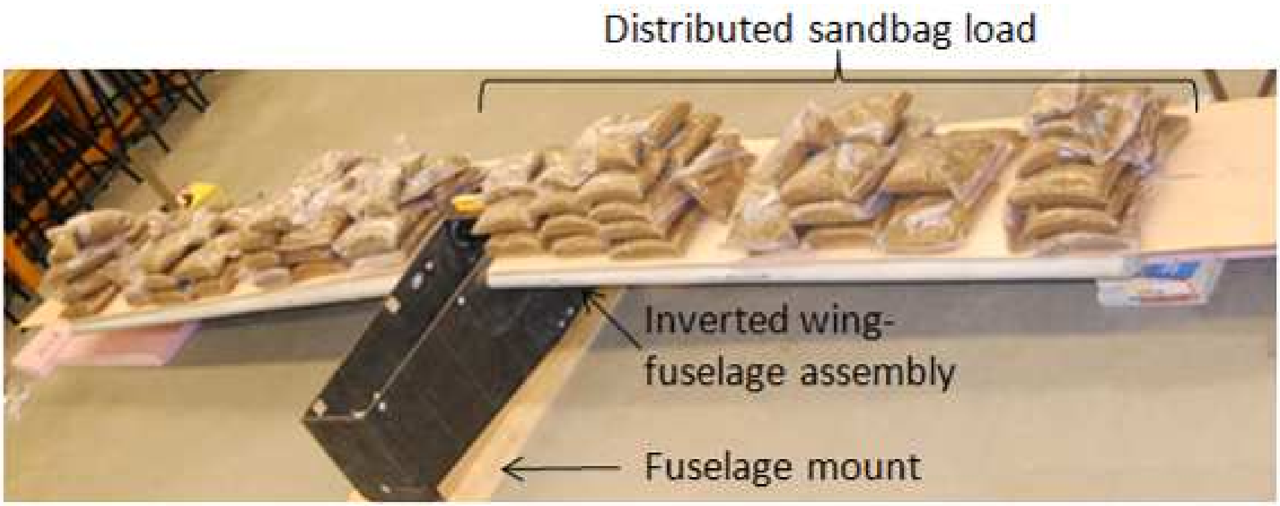


Figure D-58: Wing-fuselage structural testing

3.8 Wing Weights

Based on actual wing weight measurements taken this year, the starboard and port wings were determined to be 19.1 lb each. This results in a total weight of 38.2 lb, which is over 35% higher than previously estimated and exceeds the target GeoSurv II design weight.

3.8.1 Weight Breakdown

Figure D-59 provides a breakdown of the wing weight measurements: (A detailed breakdown is available in DR-107-18).

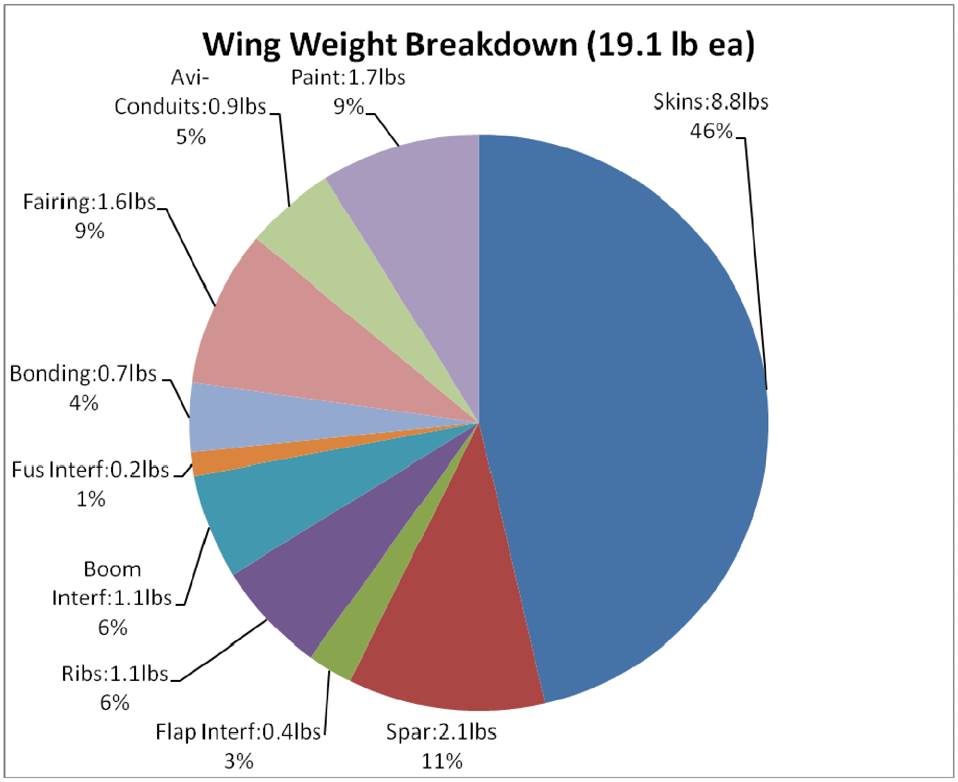


Figure D-59: Wing weight breakdown

3.6.2 Weight Reduction

As demonstrated in the wing breakdown the majority of the wing components are carbon fibre - epoxy composites, most significantly the skins. Therefore the most significant weight reduction impact would be a change to the matrix or reinforcement material. Previous work has suggested that many structural components are overdesigned using the present fabric type and recommendations indicate that an alternative reinforcement could reduce the weight by up to half. Few component configurations can be altered on the prototype wing; however minor changes could be made such as, lightening holes to the boom cradles. Other minor differences that can add up, and thus should be minimized are excess fabric, epoxy, and bonding adhesive.

4. EMPENNAGE DESIGN, FABRICATION, ASSEMBLY & TESTING

4.1 Empennage Design

The empennage employs a U-tail design consisting of two vertical stabilizers and one horizontal stabilizer, as shown in Figure D-60. Empennage control surfaces include two rudders, one on each vertical stabilizer, and two side-by-side elevators on the horizontal stabilizer. Each control surface is actuated by two servos. Details of the control system design are described in Section 5.1**.** Dimensions of the empennage are shown in Figure D-61.

Figure D-60: Prototype empennage

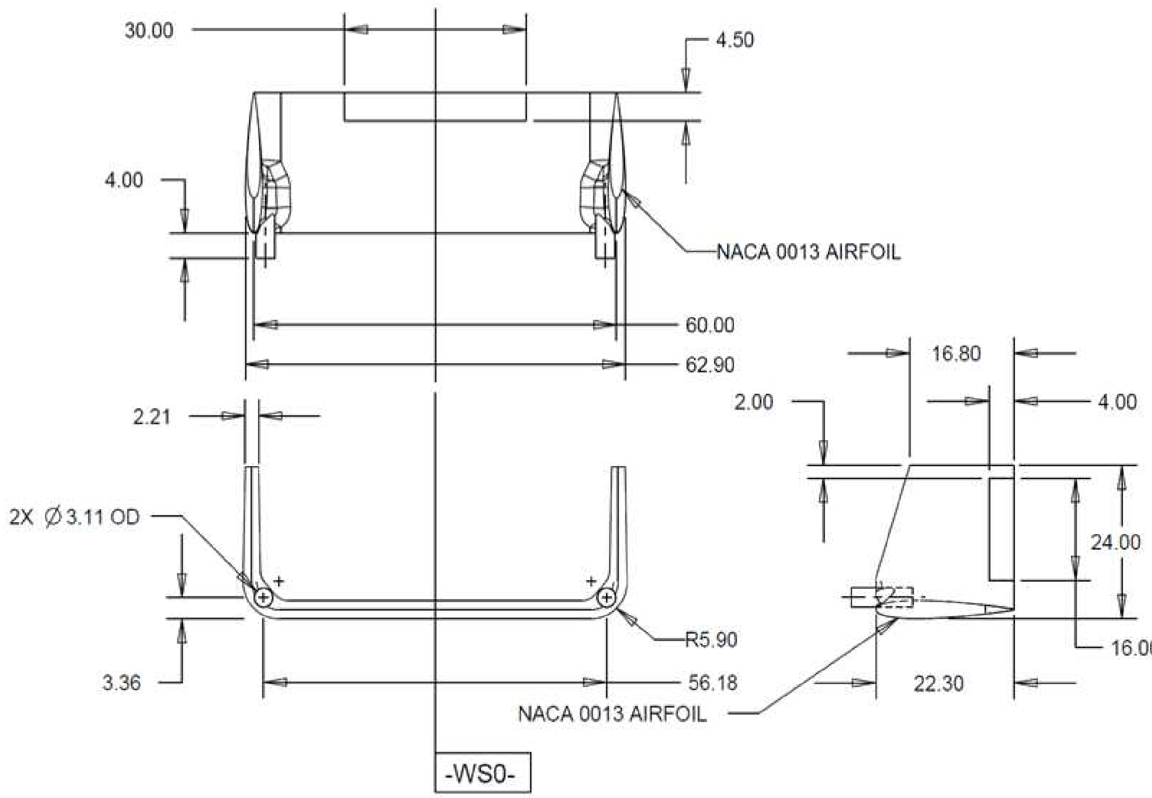


Figure D-61: Empennage dimensions

The empennage is a composite sandwich structure, with a foam core and a two-ply carbon fibre and epoxy skin. The sandwich structure provides sufficient structural properties with low weight. Lightening holes in the foam core provide weight reduction, as shown in Figure D-62.

Figure D-62: Cross section of horizontal stabilizer foam core showing lightening holes

4.2 Empennage Manufacture

4.2.1 Overview

The empennage was manufactured from blue expanded polystyrene construction foam and the same carbon fibre fabric and epoxy as used in the manufacturing of the wing and fuselage components. The majority of the empennage was completed last year; this included CNC wire cutting of all foam cores, bonding the foam cores together, manufacturing and installing the boom mounts, and applying the carbon fibre skin. Details of the manufacturing process can be found in DR87-41. The state of the empennage at the beginning of the 2008-09 year is shown in Figure D-63. Work remaining on the empennage included completion of the vertical stabilizer tips, cutting the control surfaces from the empennage structure, installing control system components, and installing the tail boom to empennage interface components.

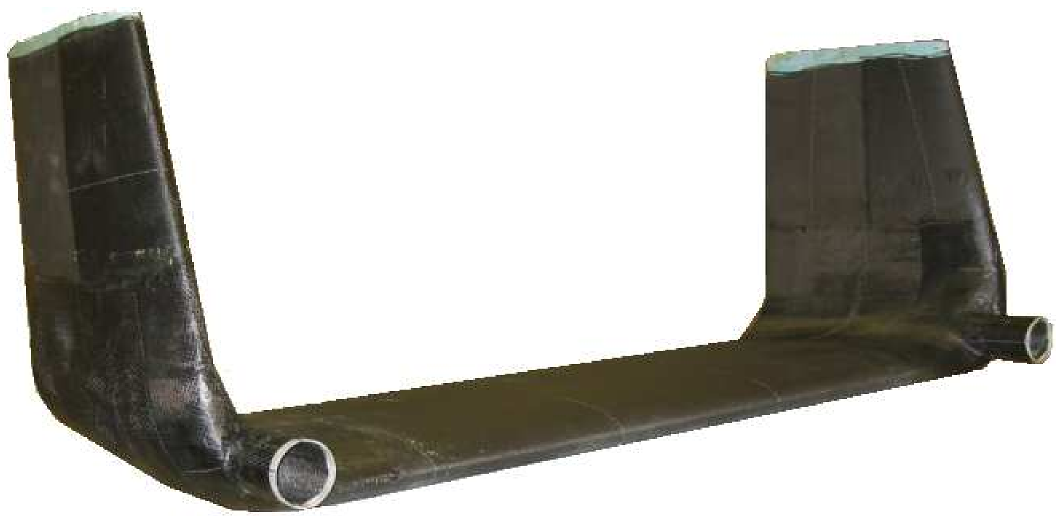


Figure D-63: Empennage at the beginning of the 2008-09 academic year

4.2.2 Completion of vertical stabilizer tips

The vertical stabilizers were completed by cutting two thin pieces of foam and sanding them to match the airfoil shape of the vertical stabilizer and to have smoothly contoured upper surfaces. The foam caps were then bonded to the vertical stabilizer tips using West System 105 Epoxy and Hardener. These caps seal the exposed lightening holes and provide a smooth stabilizer tip. Two plies of carbon fibre were then applied over the foam caps using wet lay-up, vacuum bagged, and allowed to cure at room temperature for 24 hours. Further details on this process can be found in DR107-29.

(a) (b)

Figure D-64: (a) Foam cap bonded to vertical stabilizer tip (b) Completed vertical stabilizer tip

4.2.3 Control surface cut-out and control system installation

The design of and manufacture of the empennage was carried out that each of the control surfaces needed to be cut-out from the completed structure. Each control surface was cut from the structure using a Dremel rotary tool with a diamond-coated cutting disk and a hot-wire foam cutter. An additional 1/8” (3.5mm) of material was removed around each cut-out to provide a clearance gap for each control surface. Once the control surfaces were removed, the wood hinge mounting blocks were installed. This is described in more detail in DR107-28. Cutting the control surfaces from the completed structure presented an issue of delamination of the carbon fibre skin from the foam core in the cut-out areas. To prevent delamination, the exposed foam around the cut-outs and the control surfaces was covered with carbon fibre skin using wet lay-up method. This also adds reinforcement around the hinge mounting blocks.

Cut-outs were made in the upper surface of the horizontal stabilizer and inside surfaces of the vertical stabilizers to accommodate the servos. Along the forward and aft sides of each cut-out, a slot was cut in the foam and maple blocks were bonded to the inside surface of the carbon fibre skin. These serve as mounting rails for mounting the servos. Details regarding servo mounting can be found in DR 107-30. Servos, hinges, control surfaces, and linkages are currently ready for installation following finishing and painting of the empennage.

4.3 Tail booms

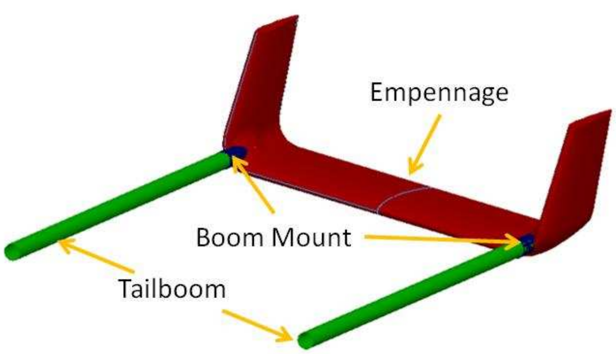
4.3.1 Overview

Each tail boom measures 8 ft in length and their outer diameter is 2.9 inches. The tail booms were manufactured using 3 plies of carbon fibre. There is an additional doubler ply that extends inward 12 inches from each end of the boom to reinforce the fastener hole locations in the booms. The manufacturing of the tail booms was outsourced to Competition Composites Inc, which is a local composite manufacturing company. Three booms were ordered, two for the prototype, and one for mechanical testing.

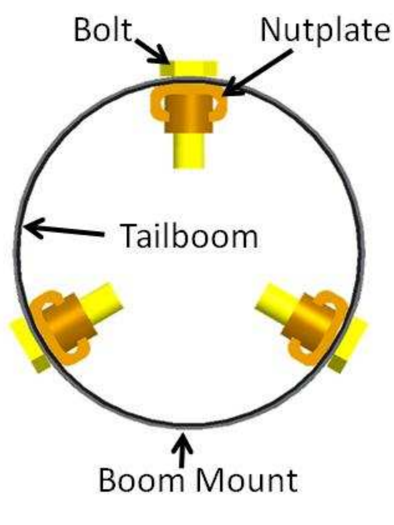
Upon receiving the tail booms, it was evident that there was significant resin starvation at multiple locations on all three booms. The manufacturer was contacted regarding this concern, and guaranteed the booms would perform as expected. Section 5.3 outlines the mechanical testing done to verify the performance of the tail booms. The design and testing of the tail booms is described in detail in DR 107-21 [5].

4.3.2 Tail Boom to Empennage Interface

Each tail boom is fastened to the empennage via three bolts that pass through both the empennage boom mount and the tail boom and fasten to a nut plate that is bonded to the inner surface of the tail boom (Figure D-65). The bolts are glass-fibre reinforced polyetheretherketone (PEEK), which makes them lightweight and non-metallic. The tail booms extend into the boom mounts 9.5 inches, and the fasteners are located 6.5 inches from the end of the tail boom. Further details of the tail boom to empennage interface are described in DR 107-20.



(a)



(b)

Figure D-65: (a) The tail boom/empennage configuration (b) The bolt and nut plate installation

4.3.3 Testing

A static load test was conducted on one tail boom to verify its performance in bending, as shown in Figure D-66. The two requirements for the tail booms are: (i) a maximum deflection angle of 2°, and (ii) an ultimate bending load of 66 lbs. This test was critical in determining whether the tail booms have sufficient strength for the GeoSurv II prototype, and to alleviate the concerns of reduced strength due to possible resin starvation. Details of the test can be found in DR 107-21.

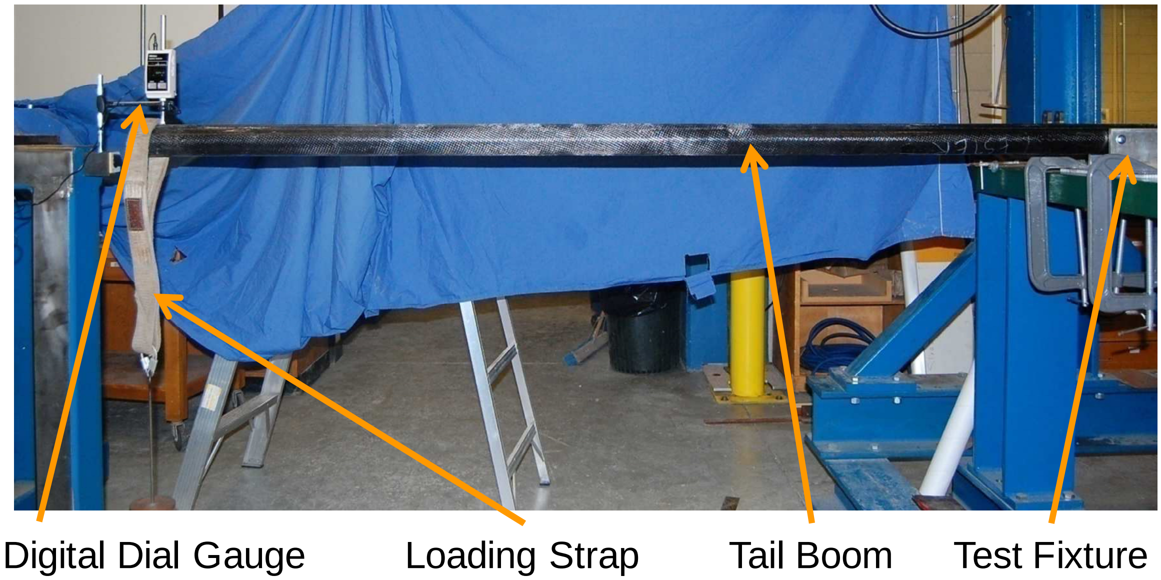


Figure D-66: Boom static load test setup

Results of the test indicated that the maximum deflection requirement of the tail boom was not met, since a deflection angle of 7.8° was reached at ultimate load. However, the bending strength requirement was met, since the tail boom supported the ultimate bending load for approximately 20 seconds before failing locally at the test fixture shear bolt locations. It was decided that these tail booms would be sufficient for the prototype aircraft, as it is unlikely to experience load factors greater than 1.5-2, where the deflection angle is within the 2° deflection requirement.

4.4 Wiring

Only servo wires will be present in the prototype empennage, therefore wiring of the empennage is simple. Servo wires are run through the central lightening holes in both the horizontal and vertical stabilizers to the boom mounts, where they connect to wire connections in the booms. The port rudder and elevator servo wires are run through the port boom, and starboard rudder and elevator servo wires are run through the starboard boom. There will be wire connections at each end of the booms and at each servo to allow for disassembly of the UAV and removal of the servos.

4.5 Weight and Balance

The as-built weight of the empennage is 13.68 lb; however this does not include paint. Paint is expected to add approximately 0.50 lb. A weight breakdown for the empennage is shown in Figure D-67. The empennage was estimated to weigh less than 10 lbs; therefore the current empennage is overweight. It is clear that a weight reduction study should be carried out on the empennage by next year's team. Some recommendations for reducing the empennage weight are described in the next section.

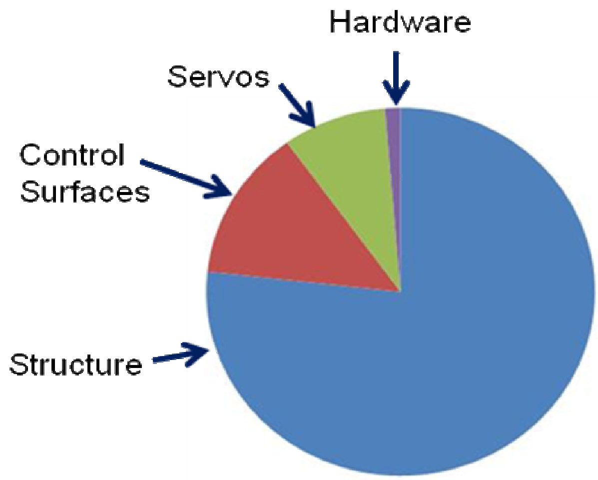


Figure D-67: Weight breakdown of empennage

4.6 Recommendations

This year's empennage sub-group had initially planned to manufacture a new empennage for the GeoSurv II prototype; however, due to time constraints this was not feasible. Therefore, the current empennage was completed as designed in order to meet the proposed first flight date. Recommendations for improving the empennage design and manufacturing method are described below. For further details, see DR 97-12.

4.6.1 Manufacturing method improvements

There are several improvements that can be made to the manufacturing method in order to reduce manufacturing time and increase overall part quality. First, by incorporating control surface cut-outs into the foam core and manufacturing the control surfaces separately, rather than cutting the control surfaces from the completed structure, would eliminate the need to cover the exposed foam with carbon fibre to prevent delamination. This would save a considerable amount of time, improve surface finish and quality, and save weight by eliminating redundant overlapping material.

Next, the vacuum bagging procedure for infusing the stabilizers should be modified to maintain ambient pressure within the lightening holes to prevent crushing of the foam cores. During manufacture, the foam core in a vertical stabilizer cracked under pressure loads and required repair.

Finally, the use of a vacuum bagging method called Closed Cavity Bag Moulding (CCBM) should be investigated. CCBM uses a custom-made, re-useable silicone vacuum bag which can be moulded to the empennage foam cores to provide a good surface finish. Additionally, using CCBM may allow the carbon fibre skin of the entire empennage to be infused at once, significantly reducing manufacturing time.

4.6.2 Material recommendations

Additional weight savings may be possible by replacing the construction foam with structural-grade. This would allow less foam to be used to achieve the same structural strength and rigidity, thus reducing structural weight. Additionally, the stronger foam would not need to be strengthened with epoxy prior to infusion of the carbon fibre skin, as was done with the current empennage. This could save as much as 1.5 lb in weight.

4.6.3 Incidence angle and control surface sizing

There was a great deal of discussion regarding the empennage incidence angle and control surface sizes of the current empennage. Based on flight tests, these parameters should be modified as required and incorporated into a second empennage. A method for adjusting the incidence angle of the empennage should also be investigated.

4.6.4 V-tail design

A study was performed by last year's team into an inverted V-tail design to replace the current U-tail. Savings of as much as 2 lb were predicted; however this did not account for the reduction in weight of servos, hinges, and control linkages. Additional details are given in DR 77-01. Substantial weight savings are possible with a V-tail design; however additional analysis would need to be undertaken by Aerodynamics Group regarding performance. It is recommended that this study be revisited by next year's team.

5. CONTROLS DESIGN, FABRICATION, ASSEMBLY & TESTING

The Flight Controls sub-group was responsible for the design, manufacture, and testing of the actuation systems to drive the GeoSurv II prototype's flight controls: the flaperons, the rudders, and the elevators. The design must meet the magnetic signature level requirements outlined in the Systems Requirements Document. This section of the final report details the empennage and wing control system interface (or hinge) designs, their respective control system designs, and the manufacture of these components.

5.1 Empennage Flight Control Surfaces Design

The 2007-2008 UAV Project Team determined the dimensions and the locations of the empennage flight control surfaces: the elevator and rudder. Further aerodynamic analysis was completed this year to verify the size and locations. In response to a request from the Avionics group, it was decided to the split the elevator into two equally sized halves. It was recommended that the Prototype be configured with unique left side and right side for control system redundancy purposes, where each side is controlled by a different receiver. Thus, the left-hand receiver controls the left flaperon, left rudder, and left elevator, while the right-hand receiver controls the right flaperon, right rudder, and right elevator.

The rudder and elevator actuation systems are subject to a number of requirements defined in the SRD. The requirements are that the actuation system must be able to deflect the elevators and rudders to +/- 30° from the neutral position. The actuation system must be able to withstand the hinge moments of 614 oz-in for the elevators and 287 oz-in for the rudders which are calculated in DR 83-12. The control surfaces and respective actuation systems must also be removable, lightweight, and contain a minimum amount of ferrous metals.

5.1.1 Empennage Control Surface Interface Design

The elevators and rudders are hinged using a continuous plastic hinge made of a modified polyolefin copolymer. According the manufacturer, the hinges have been “flex-tested for millions of cycles”, have a working temperature range from -40°F to 180°F and are UV resistant. The hinge fastens into slotted SPF (Spruce-Pine-Fir) inserts that are secondary bonded to cavities inside the stabilizer trailing edge and the control surface leading edge. Material is removed from the leading edge of the control surface to allow for the required deflection angles. This type of design results in a sealed hinge line. A sealed hinge line offers improved aerodynamic efficiency because there are no gaps in the hinge to allow flow leakage between the upper and lower surfaces. The control surfaces are installed with stainless steel machine screws that go through the carbon fibre skin, the plastic hinge, and then fasten into the SPF inserts. The exposed foam on the empennage and control surfaces is closed up using the wet lay-up method. The design is discussed in further detail in DR 97-06 and DR107-26. Figures 71 and 72 show cross-sections of the design.

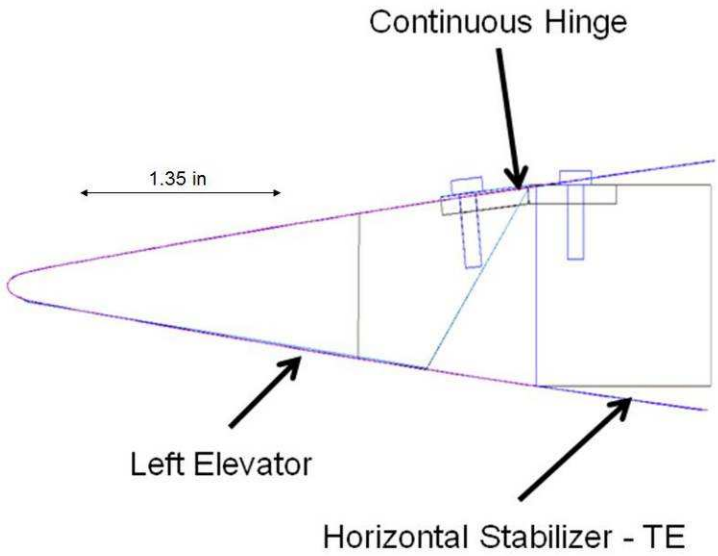


Figure D-68: Left Elevator Interface Detail

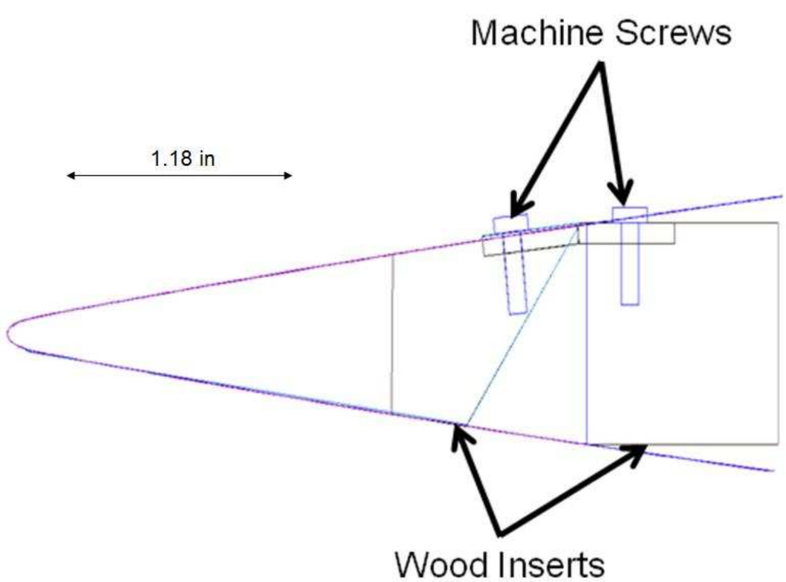


Figure D-69: Left Elevator Interface Detail

5.1.2 Manufacture of Elevators and Rudders

The elevators and rudders were cut directly from the empennage. This created the cut-outs needed for the control surfaces and saved manufacturing time by using the cut-out parts. The carbon fibre skins on the lower surfaces were cut with a Dremel rotary tool, and the foam core was hollowed out using a hot wire foam cutter. The SPF inserts were cut to shape using various woodworking tools such as a table saw, band saw, and chisels. The inserts were bonded into the leading edges of the control surfaces using secondary bonding. Then, carbon fibre was laid up around the exposed foam and wood and vacuum pressure was using to pull the carbon fibre flush against the surfaces. See DR 107-28 for more information. Photographs of the completed elevators appear in Figure D-73.

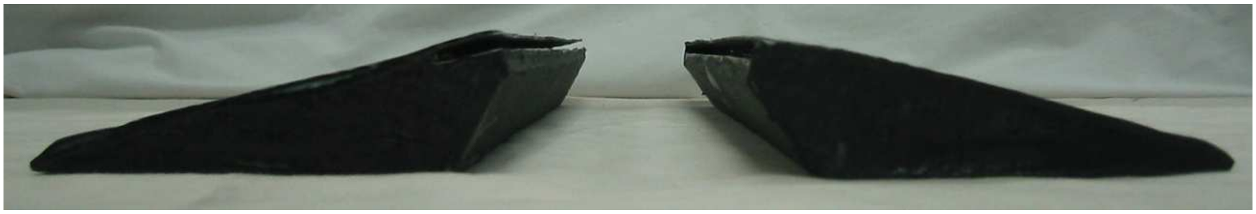


Figure D-70 – Completed Elevators

5.1.3 Manufacture and Installation of Empennage Wood Inserts

The SPF inserts were cut to shape using various woodworking tools such as a table saw, band saw, and chisels. Lightening holes were drilled into the inserts to reduce weight. An X-Acto knife was used to create the insert cavities in the foam in the empennage trailing edges. The inserts were bonded into the trailing edges of the empennage using secondary bonding. Then, carbon fibre was laid up around the exposed foam and wood and vacuum pressure was used to pull the carbon fibre flush against the surfaces.

5.1.4 Empennage Control System Overview

The control surfaces on the empennage are actuated using radio control servos actuators. There are two servos per empennage control surface. Two servos are necessary in order to withstand the aerodynamic loads imposed on the control surfaces during certain manoeuvres, and to allow for redundancy in the case of a single servo failure. Although the remaining functional servo would not be able to withstand the maximum aerodynamic loads, it will provide the pilot with enough control to safely land the aircraft. Moreover, if an entire control surface failed, there is another corresponding control surface that will provide the pilot with limited control.

The servo used to actuate the control systems is the Hitec HSR-5990TG robot servo. Its selection and particular issues are discussed in DR97-09.

In terms of the empennage control system, the linkages consist of a titanium pushrod that has ball links at either ends: the servo arm and control horn. The servos are mounted to maple rails that have been secondary bonded into pockets underneath the empennage skin with socket head screws. Aluminum servo arms are screwed onto the servo output shaft. Heavy duty nylon ball links attach to the servo arms with nuts and bolts. The titanium pushrods screw into the ball links. The titanium pushrods have a distinct left hand thread at one end, and a right hand thread at the other, similar to a turnbuckle. This means that they cannot completely unscrew themselves from the ball links. Another ball link attaches to the other end of the pushrods. These ball links are bolted to chrome plated heat treated steel control horns, which are screwed into the SPF control surface inserts. See DR 97-21 for more information. Figure D-74 depicts the empennage flight control system.

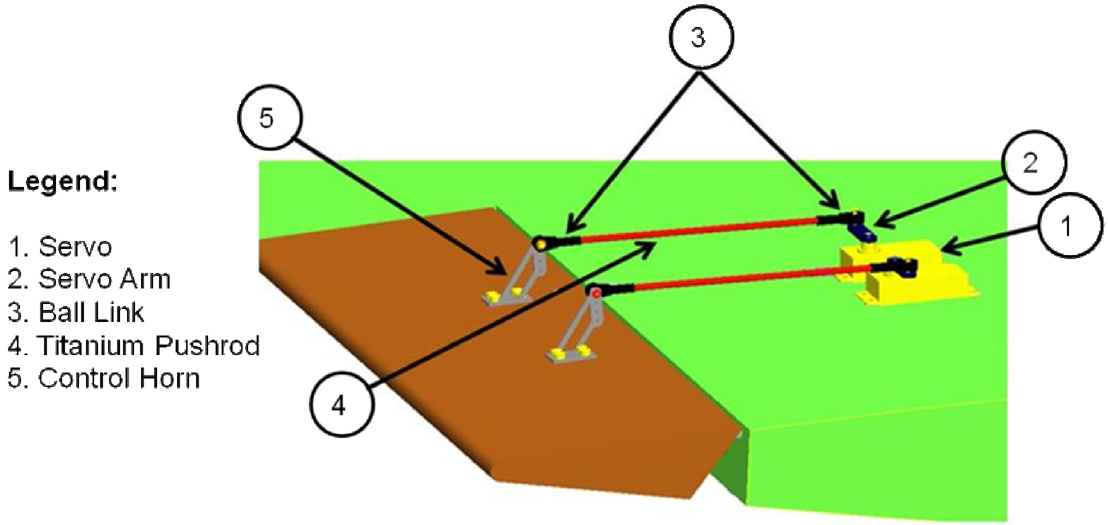


Figure D-71: Control System Diagram

5.2 Empennage Mock-Up

The 2007-2008 empennage sub-group manufactured a test section to test and troubleshoot the VARTM manufacturing process as well as perform some structural testing. The current year's Flight Controls sub-group used this section to test and troubleshoot the control surface interface and control system design. The hinges and linkages were installed on the test section and the control system was tested. Tests proved that the elevators and rudders can deflect to +/-30° from the neutral position and that if one servo fails the other has the power to drive the control surface. Working with the empennage mock-up also provided invaluable manufacturing lessons such as how to cut and shape the SPF inserts, and how to manufacture and machine the carbon fibre composite. See DR 107-27 for more information. Figure D-75 shows photographs of the top and side view of the empennage mock-up.

Top View

Side View

Figure D-72: Empennage Mock-Up Photos

5.3 Flaperon Control Surface Design

During the 2008-2009 year, the main focus of the flight controls subgroup was to complete manufacturing of the control surfaces in time for first flight. Although some design had been done for the flaperon control surfaces in previous years, much detail work still needed to be completed and manufacturing had not yet begun.

Throughout the fall term, the design work was completed for the flaperon hinge and servos were selected for use on all of the prototype control surfaces. Final designs for the flaperon interface and the actuation system are discussed in sections 5.3.1 and 5.3.3, respectively. Additional information can also be found in DR 97-09 and DR 97-20.

The winter term consisted mostly of manufacturing work, with two flaperons having been completed. The details of the work done are given in section 5.3.2 and also in DR 97-20. Several problems arose throughout the design and manufacturing process; these are discussed in further detail in the upcoming sections.

At this point, the Flight Controls subgroup is satisfied with the work achieved this year; the main goal of first flight is within reach. However, there are several areas which could be improved; some recommendations for future teams will be presented later in the report.

5.3.1 Flaperon Control Surface Interface Design

The flaperons on the GeoSurv II are made from a carbon/epoxy foam core sandwich structure. There is a carbon fibre spar tube running along the leading edge of the flaperon, which acts as the hinge line for actuation. An exploded view of the flaperon and the hinge components can be seen in Figure D-73, below.

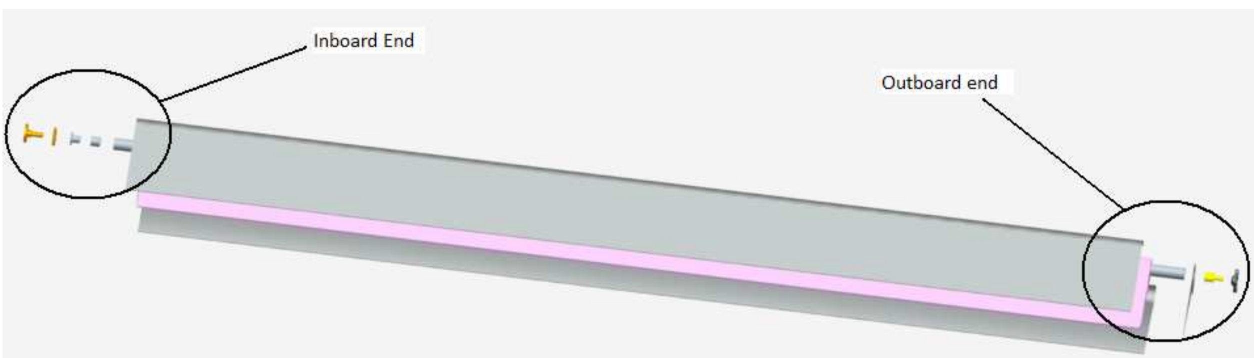


Figure D-73: Flaperon exploded view

This figure highlights the inboard and outboard ends of the flaperon. These are the two areas where the flaperon is hinged to the wing. The components are seen in cross-section in Figure D-74 and Figure D-75 below.

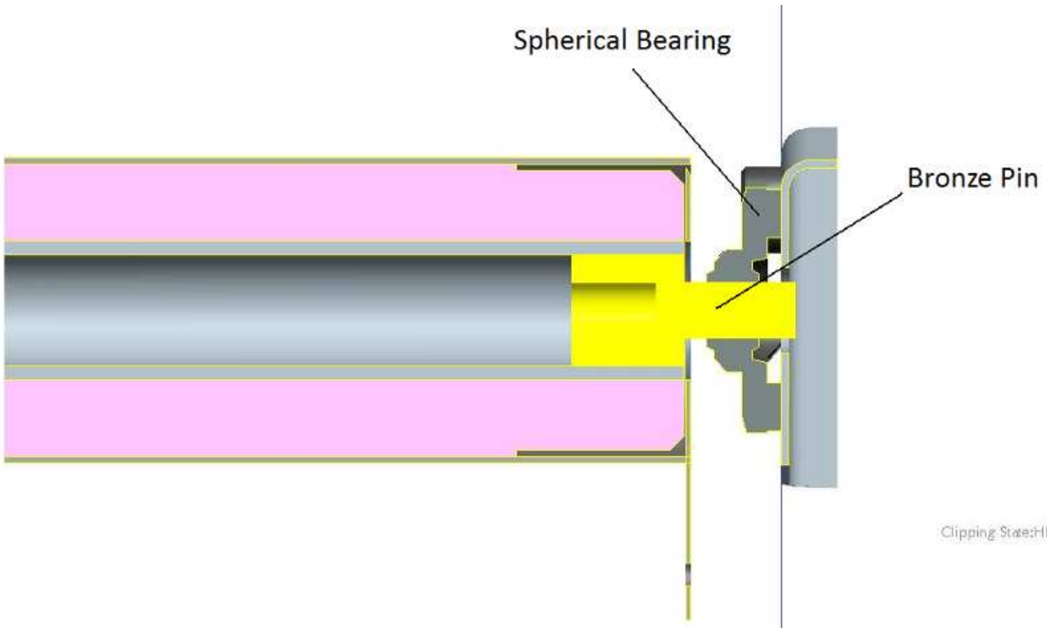


Figure D-74: Inboard Hinge Components

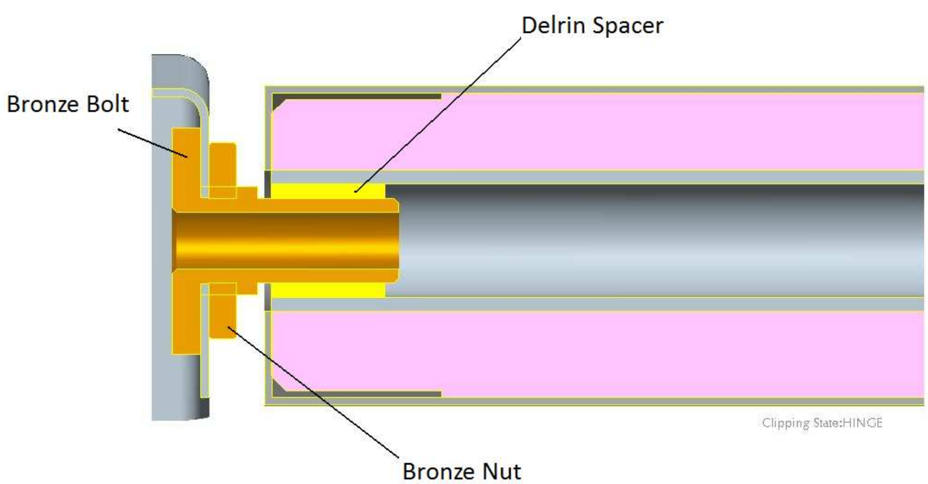


Figure D-75: Outboard Hinge Components

At the inboard end, there is a bronze pin permanently bonded inside the spar tube. A plain spherical bearing is mounted on the wing trailing edge cap, with a titanium nut plate bonded to the opposite side of the cap to retain the bolts. This bearing, purchased from Igus, is shown in more detail in Figure D-76 below.

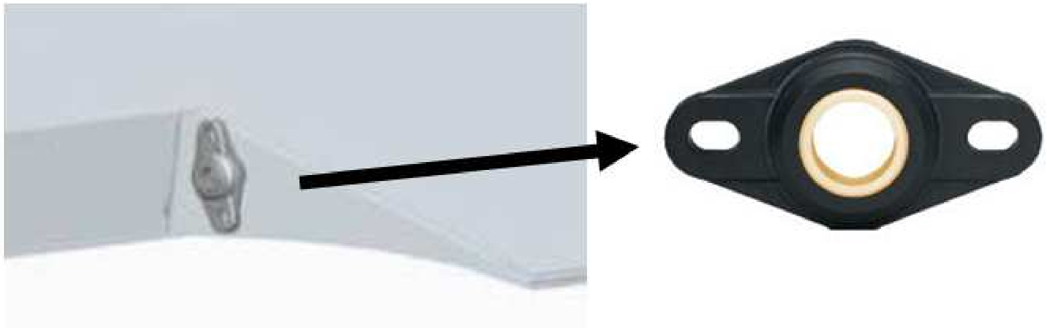


Figure D-76: Igus spherical bearing

At the outboard end, there is a Delrin spacer permanently bonded inside the spar tube. This spacer bridges the gap between the inside diameter of the tube and the outside diameter of a flanged iglide sleeve bearing, purchased from Igus. A flanged bronze bolt is inserted through a hole in the trailing edge cap and is retained by a bronze nut permanently bonded to the opposite side of the cap, as shown in Figure D-75 above.

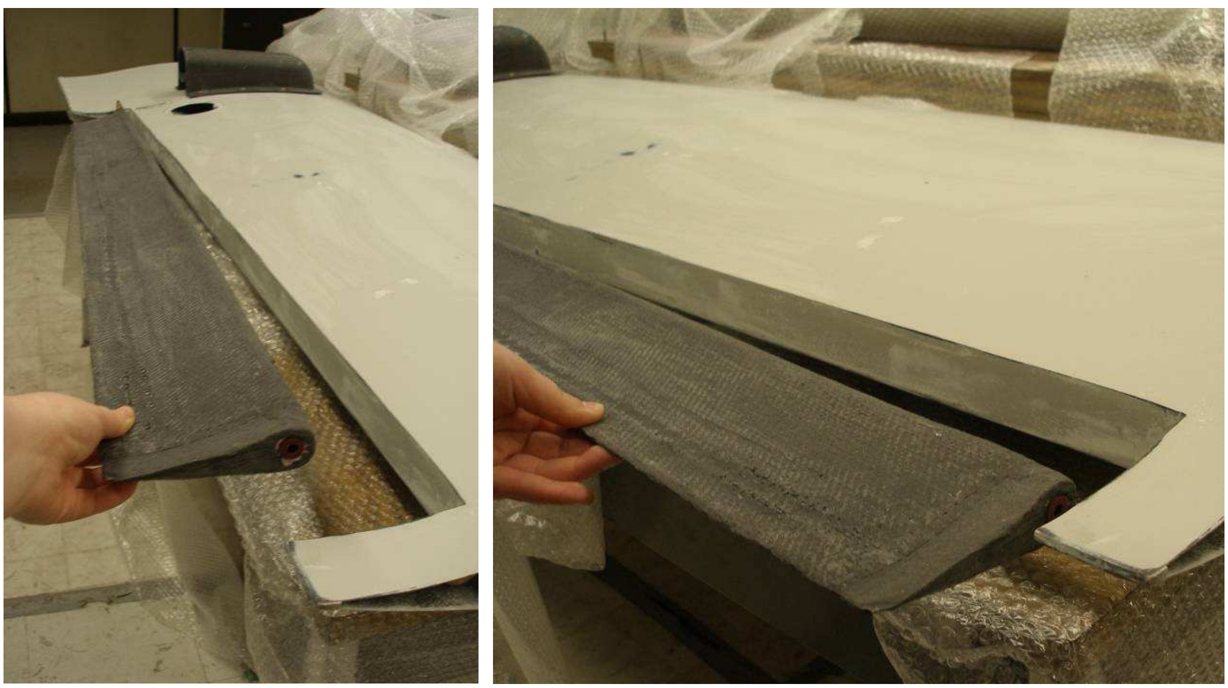


Figure D-77: Flaperon Installation

Installation and removal of the flaperons is a simple process. To install, the inboard pin is first inserted into the spherical bearing at an angle; this is shown in Figure D-77, above. Next, the outboard end of the flaperon is aligned with the trailing edge cap, and the bronze bolt is inserted through an opening in the wing tip fairing.

5.3.2 Flaperon Manufacture

The first step in manufacturing the flaperons was to cut the foam core. This was done in two halves (upper flaperon and lower flaperon) on the CNC router table in the Carleton University MAE machine shop. The two completed halves are shown in Figure D-78 below.

Figure D-78: Upper and lower foam cores

Once the foam cores were cut, the trailing edge was extremely thin and delicate. In order to prevent degradation, the two halves were coated with a thin layer of epoxy. At the same time, the spar tube was bonded in and the two halves were bonded together. A picture of the two halves sitting together with the spar tube inside is shown below in Figure D-79.

Figure D-79: Foam core and spar tube

Next, two bellcranks were cut from a previously manufactured carbon/epoxy flat panel. The control horns were reinforced with lightweight plywood to prevent delamination and the bellcranks were bonded to the inboard ends of the flaperons. The bellcrank as bonded to the end of the flaperon is shown in Figure D-80 below.

Figure D-80: Bellcrank bonded to inboard end

A wet lay-up of two layers of carbon fibre was done on both the inboard and outboard ends of the flaperon to improve stiffness and load transfer to the skin. This lay-up was cured under vacuum pressure to ensure a smooth surface finish, as shown in Figure D-81.

Figure D-81: Wet lay-up curing under vacuum

Finally, the foam cores were wrapped with two layers of carbon fibre and then were infused using VARTM. It is believed that during the VARTM process significant amounts of epoxy leaked into the lightening hole in the foam core, vastly increasing the final weight of the flaperon.

5.3.3 Flaperon Actuation System Design

The majority of the actuation system design had been completed by previous year’s groups; however some small changes were made as problems were encountered throughout the year. The original servo mounting design as it was at the beginning of the 2008-2009 school year is shown in Figure D-82 below.

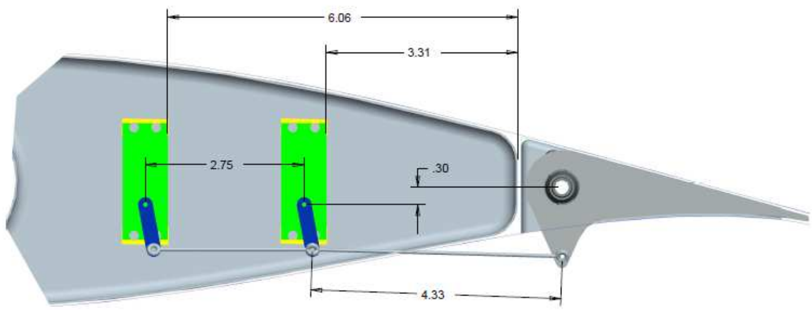


Figure D-82: Original servo mounting design and dimensions

While installing all of the components and preparing for wing close-up, it was discovered that it was extremely difficult to remove the servos through the access hole in the lower wing skin. Several UAV team members attempted to unscrew the servos from the rib, however they were unsuccessful due to the tight space and interference from the components. To address this issue, it was decided to mount the servos on a lightweight plywood plate, which would then be bolted to the wing rib. To retain the bolts, nuts were inserted into smaller plywood plates which were then permanently bonded to the opposite side of the rib. The bolts on the plywood mounting plate were able to be placed such that they are easier to remove. A picture of the final mounting configuration is shown in Figure D-83 below.

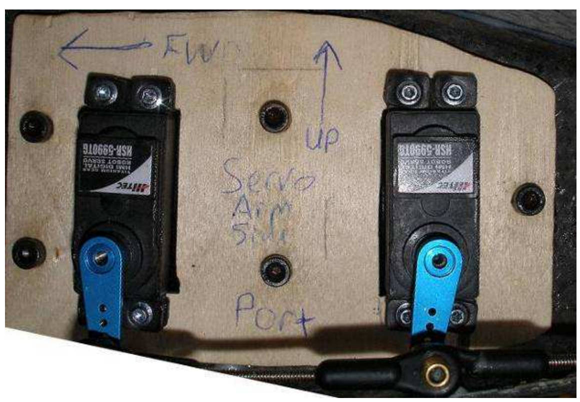


Figure D-83: Final servo mounting design

One other problem which was encountered during the manufacturing and testing process was the interference of the bellcrank with the wing close-out spar at approximately -25°. This interference limited the amount of deflection which could be achieved by the flaperons. A redesign of the bellcrank is still currently being carried out, with some minor increases in deflection having been achieved thus far.

6. LANDING GEAR DESIGN, FABRICATION, ASSEMBLY & TESTING

6.1 Landing Gear Overview

The landing gear setup on the prototype features a tripod design; the configuration of the nose and main landing gears is shown in the Figure D-below. As discussed in DR 103-01, there is height difference of 2 in between the nose (28 in) and main landing gear (26 in).

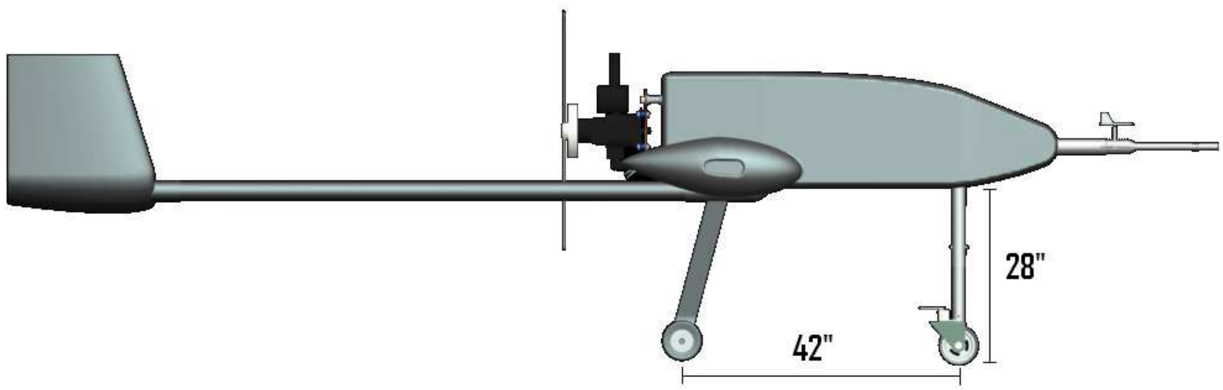


Figure D-84: Landing gear configuration on the prototype

6.2 Nose Landing Gear

In the following sections, reasons for changing the previous design are discussed; also, the various nose gear designs that were considered are presented along with the final decision on the nose strut design. The design for the shock absorber, steering mechanism and nose wheel selection are also discussed in more detail.

6.2.1 Requirements for Nose Landing Gear

The nose landing gear must adhere to the requirements as stated in the air worthiness standard. In addition, the landing gear on the GeoSurv II must comply with the System Requirements Document (SRD). The UAV standard(s) which are used throughout the design are outlined below:

Always keep a minimum propeller-ground clearance of 20% of the propeller diameter (or 7.2in)

In addition to the design guideline(s) above, the following certification requirements must be met:

- 23.303: Safety factor of 1.5 used throughout
- 23.473: Withstand sink speed between 7-10 ft/s
- 23.477: Landing gear arrangements
- 23.479: Level Landing conditions

- 23.481: Tail down landing conditions
- 23.485: Side load conditions
- 23.499: Supplementary conditions to nose wheel (please refer to FAR23)
- 23.609: Suitably resistant to corrosion, weather etc.
- 23.611: Detailed drawings for parts that require servicing (shock absorber)
- 23.623: Bearing factors must be large enough to withstand the forces due to relative motion in joints with clearances.
- 23.625: Fitting factor of at least 1.15 must be used throughout
- 23.723: Energy absorption tests must be performed to prove that the takeoff and landing weight limits will not be exceeded.

6.2.2 Design Change

Figure D-85: Last year's nose gear design concept

The reasons for changing the design of last years' team include:

- High complexity of curves in the strut geometry
- Resulting geometry requires difficult moulds to be fabricated
- New carbon fibre manufacturing methods would need to be developed, tested and implemented
- Time constraints will make this design difficult to execute.
- Redesign is inevitable due to inadequate fuselage attachment and lack of steering.

The main reason for changing last year's nose gear design stems from manufacturing difficulties. In future years, more research and development of the manufacturing process should be conducted to further reduce the weight of the aircraft by having a composite nose gear. The new design features a telescopic strut design with a rubber shock absorber.

6.2.3 Nose Strut

Aluminium was chosen for the construction of the nose gear because of its low weight, relatively easy machinability and its structural properties. The aluminum will not interfere with the magnetometer readings, which is critical in the GeoSurv II. Several iterations on the nose strut design have been performed and different designs have been considered. The main design criteria used when designing the nose strut were:

- *Manufacturing:* How difficult it is to build and assemble?
- *Serviceability:* Can critical components be accessed and serviced quickly and easily (i.e.: shock absorber)?
- *Cost:* The overall price of materials and labour

With the following constraints in mind, the final design and the layout of the attachment to the fuselage bulkhead can be seen throughout the next couple of pages:

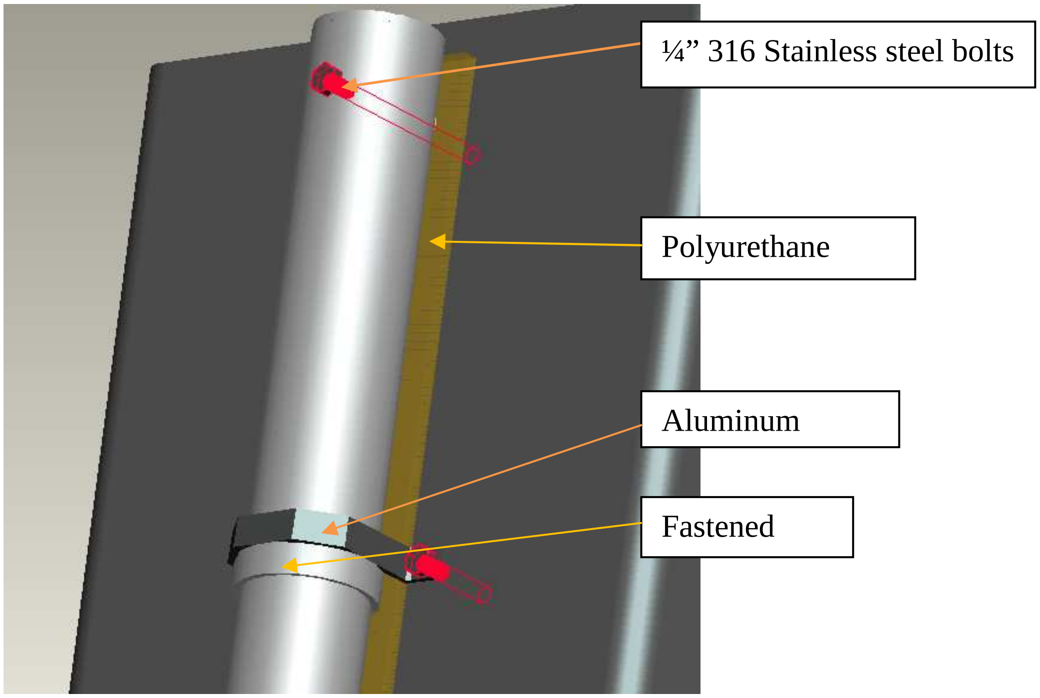


Figure D-86: Attachment configuration on the fuselage bulkhead.

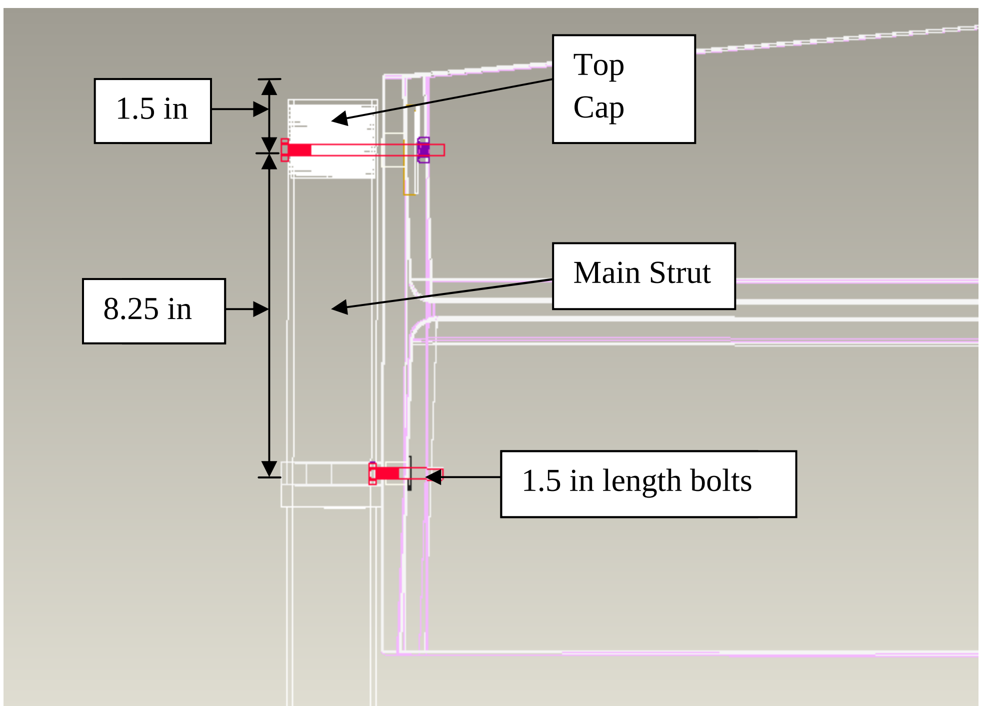


Figure D-87: Cross-Section of the attachment to the bulkhead.

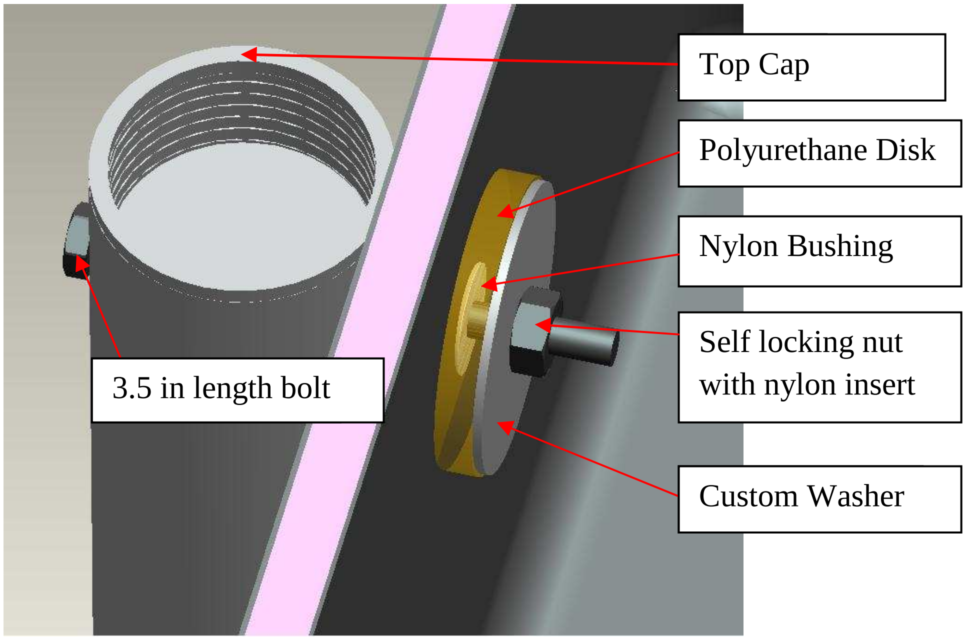


Figure D-88: Attachment to the fuselage bulkhead as viewed from inside the fuselage (upper half).

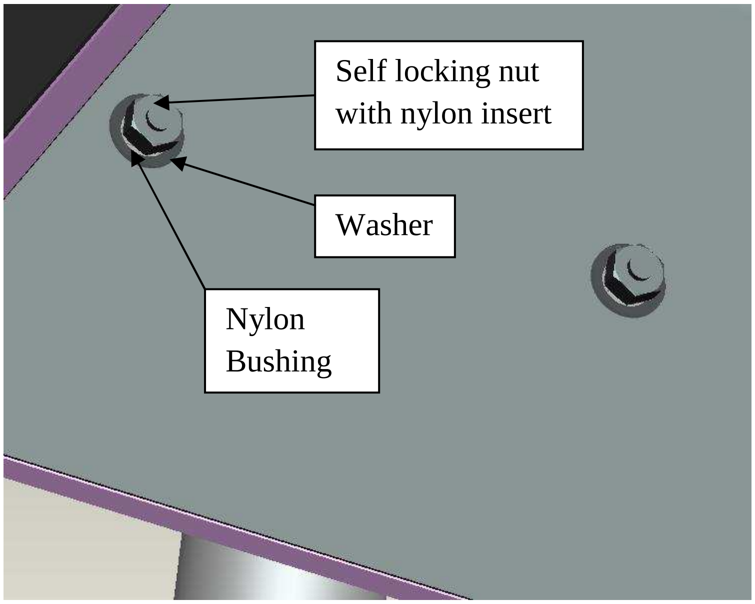


Figure D-89: Attachment to the fuselage bulkhead as viewed from inside the fuselage (bottom half).

The polyurethane backing is used to distribute the line load of the cylindrical landing gear on the outer skin of the front bulkhead. The aluminum bracket serves in resisting the bending moments due to tire friction on landing as well as transmitting the vertical landing loads through the collar into the two attachment bolts. The top cap prevents the main strut from crushing when the bolt is tightened at the top and it also completes the shock absorber structure allowing for the compression of the rubber disks. The current design allows for 19.125 in of space for the rubber compression disks. The nose gear measures 28.25 in from the runway to the base of the fuselage.

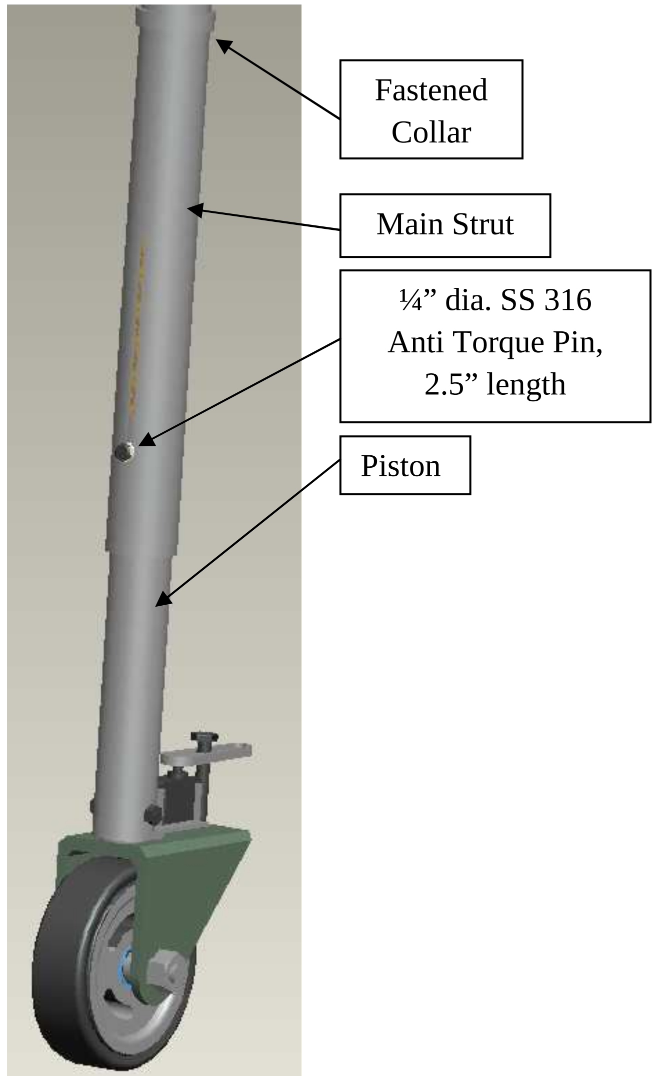


Figure D-90: Nose gear assembly

6.2.4 Shock Absorber

Many designs were considered and investigated, leaf spring, hydraulic, metal spring, but the simplest solution was found to be rubber shock absorbers. The design for the prototype employs rubber 'disks' that are stacked on top of each other to provide the damping characteristics required for landing. These polyurethane rubber disks sit in the main shock absorber tube in configuration shown in Figure D-91. The shock absorber piston is bolted to the base plate with a ¼ in, 316 stainless steel bolt.

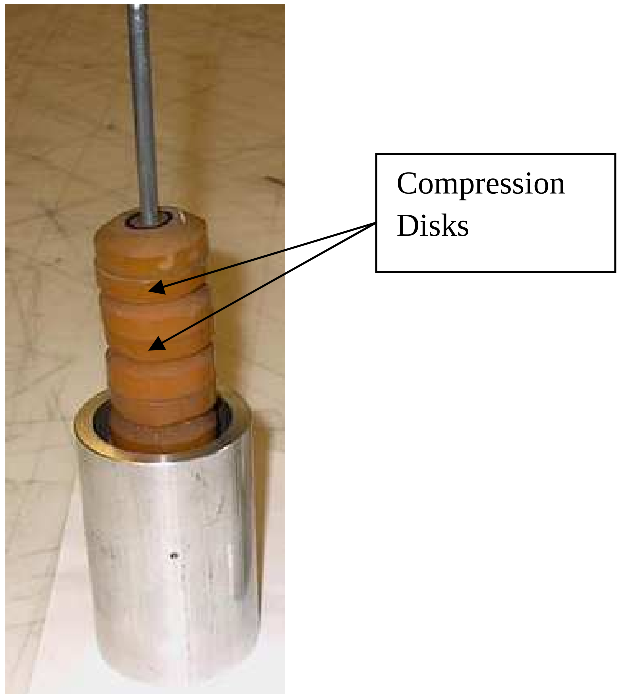


Figure D-91- Picture of the rubber disks arranged in the cylinder of the prototype shock absorber.

In the final design the rubber disks will be stacked in a similar fashion inside the shock tube. The rubber guide rod prevents the misalignment of the rubber disks inside the shock tube, and makes the overall assembly process more efficient. Standard off-the-shelf lubricant will be sufficient for minimising the friction of the rubber disks on the rubber guide rod.

The rubber disk design was modified from *[Aircraft Landing Gear Design, S.Currey P.54]* and adapted for a smaller radius of disks:

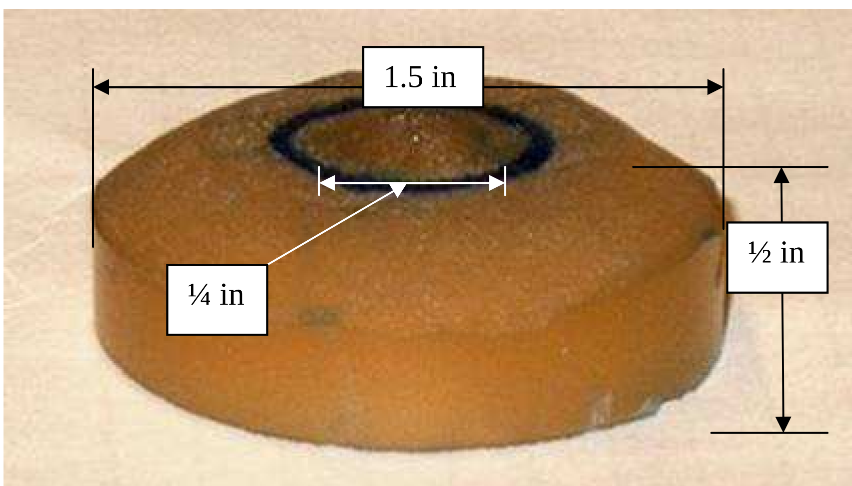


Figure D-92- Picture of a rubber compression disk used in testing.

Two polyurethane rubbers with durometer ratings of 60A and 40A were chosen to conduct testing and optimization of the shock absorber design. Some key advantages of this design over springs or hydraulic dampers, is the ability to check and inspect the disks quickly before any flight to ensure that shock absorber is not damaged in any way. In the event of a disk being cracked or faulty, maintenance can be performed and a new disk put into place resulting in no down time or flight delay. The stroke of the damper can be changed by increasing or decreasing the amount of disks in the component. The energy that needs to be absorbed is calculated by the kinetic energy:

$$\text{Energy Cap. (in.-lbs.)} = \frac{W}{772} \times V^2$$

Equation 1

For our purposes, for a 30 lb load with a sink speed of 7ft/s results in 270 in-lb of kinetic energy. The testing of the shock absorber has been completed and the results are summarized in DR 97-15.

6.2.5 Steering Mechanism

The steering will be achieved by a servo attached to the base plate of the nose gear. A shoulder bolt is placed through a slot on the servo arm and is fastened directly to the caster. The servo arm rotates the caster which in turn rotates the wheel. A thrust bearing is sandwiched between the base plate of the nose strut and the caster allowing the caster to rotate 360 degrees. The thrust bearing is slightly recessed as shown in Figure D-96; this is to reduce the thrust bearings' exposure to the environment. The servo is mounted to the base plate with two brackets in the configuration shown in Figure D-93. The brackets are brazed directly to the base plate for ease of manufacturing and taped on the top to allow for the servo to be screwed on. Rubber grommets are to be used to reduce the vibrations when taxiing and landing.

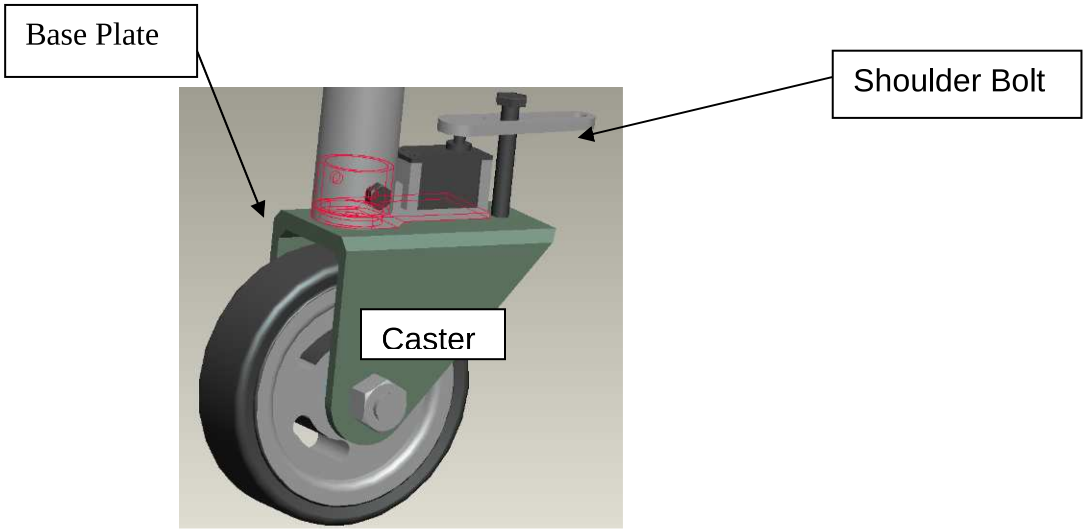


Figure D-93: Base plate assembly.

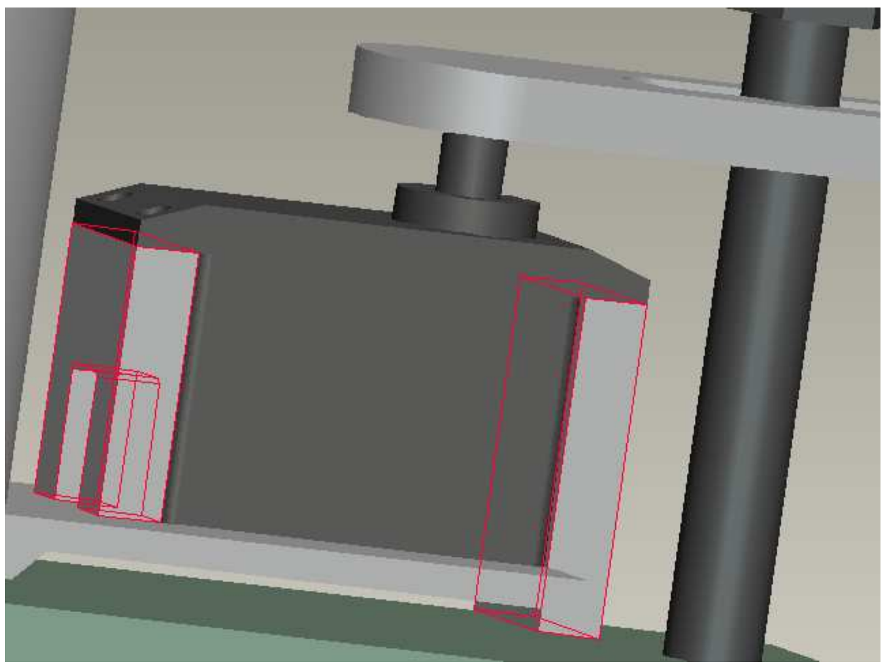


Figure D-94: The two servo support brackets are brazed directly to the base plate.

The servo arm and the caster rotate on different axes and therefore their radii of rotation are different. To solve this problem, a slotted servo arm is used to allow the caster to rotate. Figure D-95 shows the servo arm at 0° and 90° to the base plate. When the servo is 90° it corresponds to almost 45° of rotation for the caster. Only approximately 10° of rotation are required and thus the slot in the servo arm will be decreased to insure that the caster is physically restricted to this rotation.

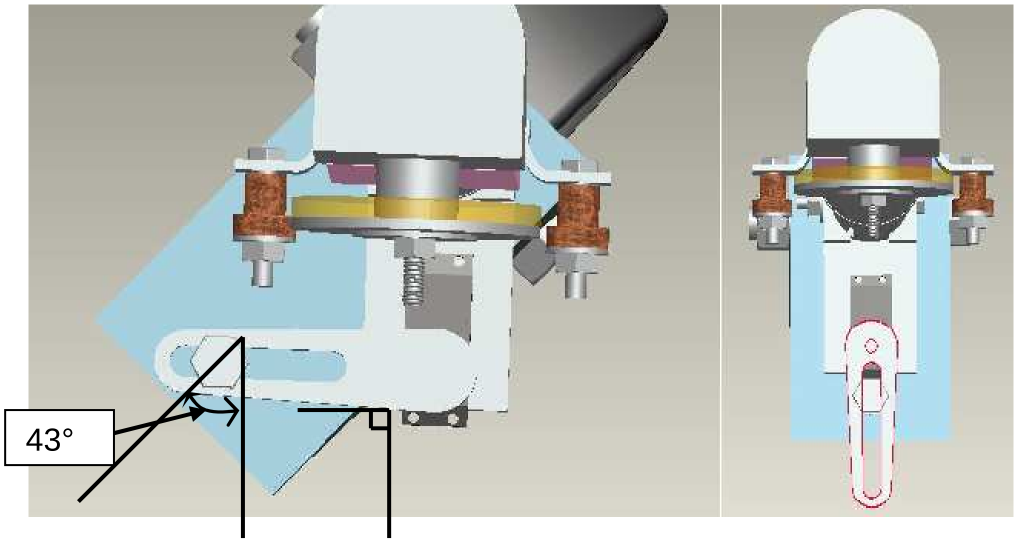


Figure D-95: The range of motion of the wheel.

The wheel is connected via a 5/8 in diameter 3.75 in long aluminum axle fastened through the caster. Two aluminum spacers are pressed up against the roller bearings to achieve a rigid wheel assembly. The wheel-caster assembly can be seen in Figure D-96.

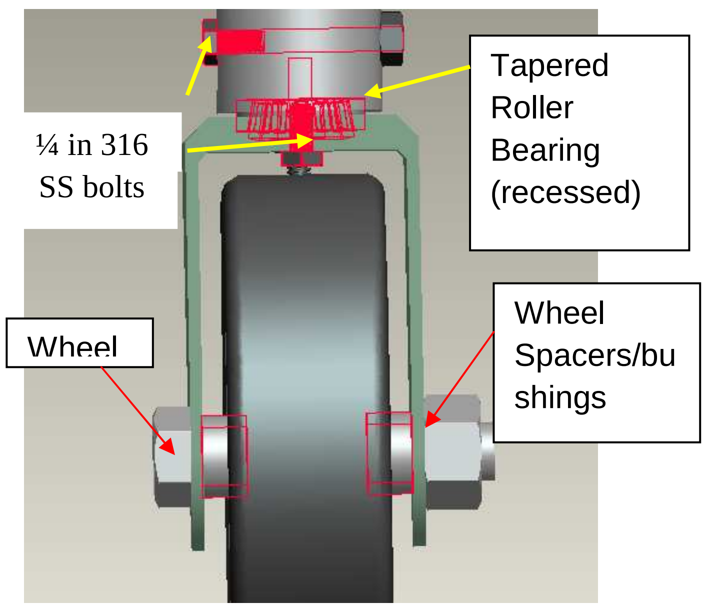


Figure D-96: Figure of the wheel assembly configuration.

6.2.6 Wheel Selection

The nose wheel will be the same as the wheels already purchased for the main landing gear. It was decided that 6 in dia. Matco pneumatic wheel assembly and tires would better suited for taxiing and absorbing energy on landing than 4 in dia. wheels. The wheel assembly specifications can be found in DR 87-10.

6.3 Composite Main Landing Gear

The carbon fibre main strut designed and analyzed the previous year are described in DR 87-11 and DR 87-37, respectively. The design is a leaf spring landing gear consisting of 32 plies of carbon fibre with a configuration of $[(0/90)_8/(\pm45)_{16}/(0/90)_8]$ and the geometry shown in Figure D-97.

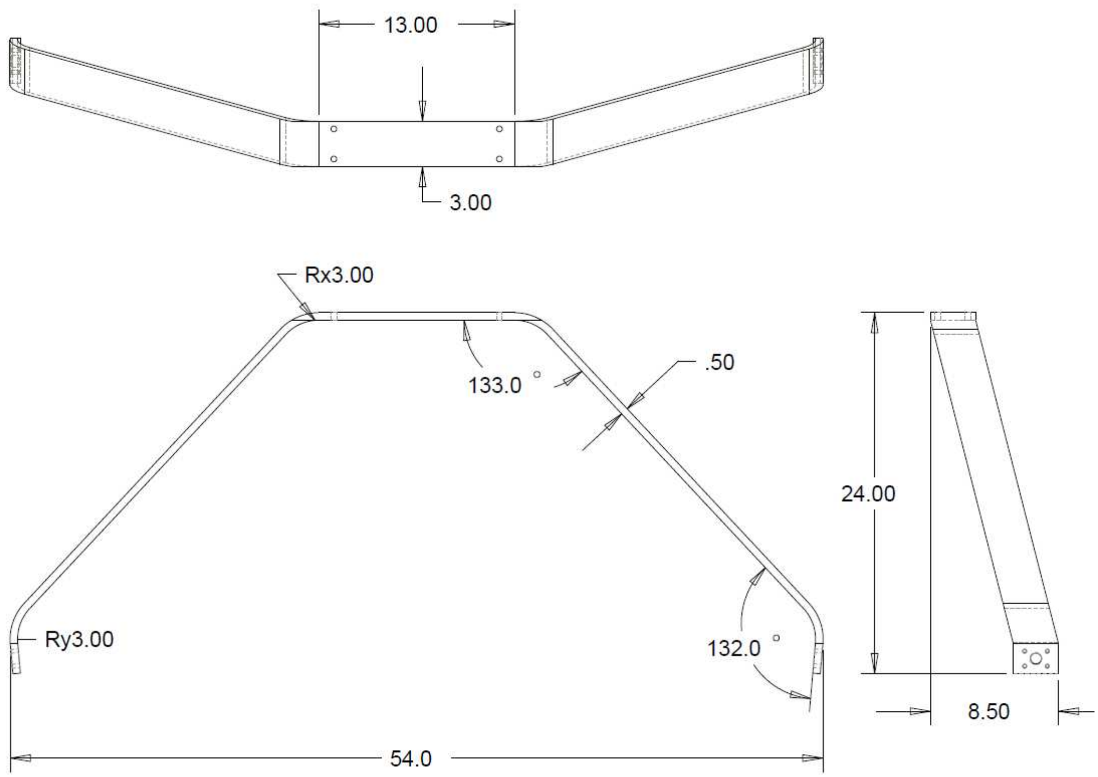


Figure D-97: Main strut dimensions in inches

6.3.1 Manufacturing Tests

The main strut manufactured last year had two major problems which compromised the structural integrity of the component: air pockets inside the structure, and surface wrinkles at the shoulders of the strut. To improve the quality of the main strut three manufacturing test were made. The first test changed the manufacturing method from a wet lay-up to a VARTM infusion in order to reduce the air pockets. After successfully reducing the air pockets inside the main strut, the second test was an infusion of a 130° bend on an aluminum mould. This test proved the fibres could be curved without producing wrinkles. The third test was a closed mould infusion using medium density fibreboard (MDF); however, this test piece did not infuse fully. The detailed methodology and results of these tests are described on DR 97-07 and the sample results are shown in

Table **D-2**.

Table D-2: Result comparison from the manufacturing tests

1st Test	2nd Test	3rd Test
• Wet lay-up • Thickness: 0.47in  	• VARTM • Thickness: 0.50in 	• VARTM • Thickness: 0.49in 

6.3.2 Manufacturing of the Main Strut

Following the manufacturing tests, the main strut was manufactured as described in DR 107-13. The manufacturing process begins with an aluminum mould bent to the shape of the main strut. This is followed by laying 32 plies of carbon fibre on the mould as shown in Figure D-98.

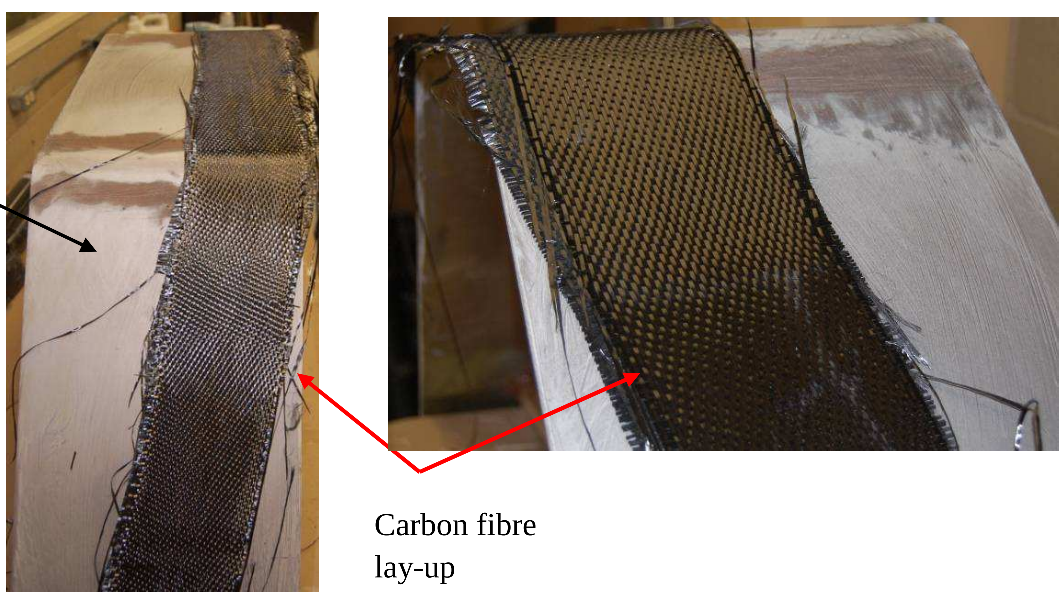


Figure D-98: Ply lay-up on the aluminum mould

Then the VARTM setup is placed as show in Figure D-99.

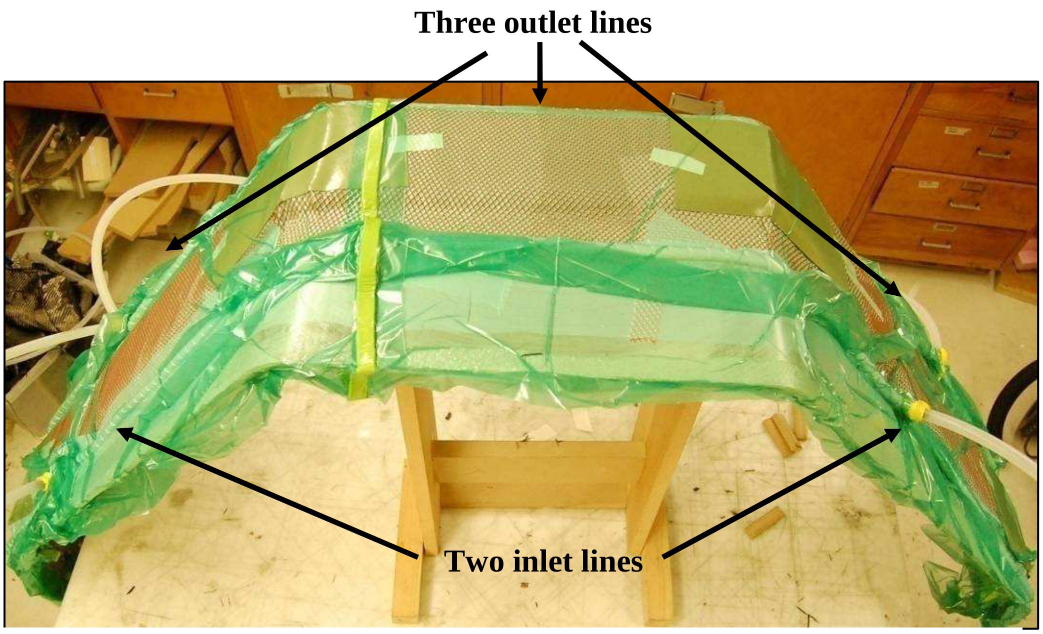


Figure D-99 – VARTM setup for the main strut

The infused strut was then trimmed and sanded. Its evolution is shown in Figure D-100.

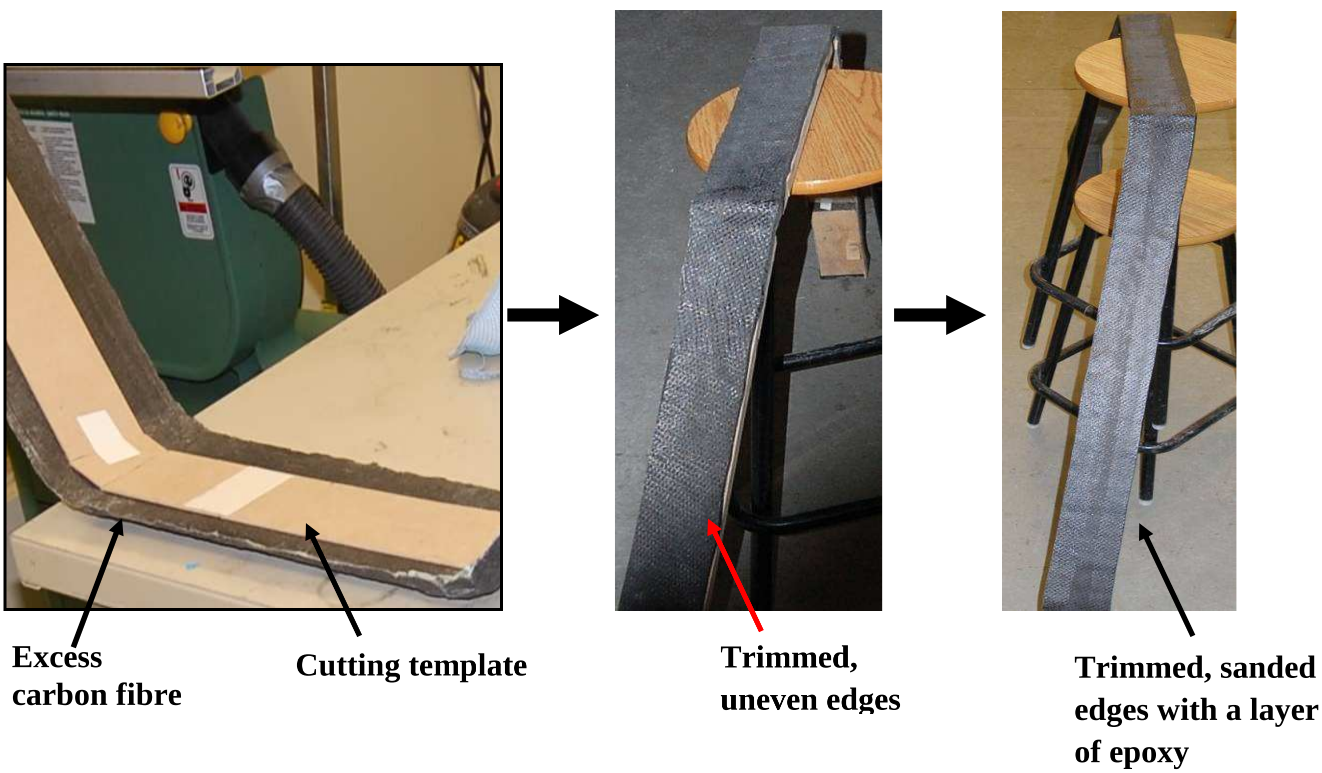


Figure D-100 – Trimming process for the main strut

6.3.3 Wheel Attachment

The 6 in diameter Matco Mfg wheel is attached to the main strut as shown in Figure D-101. The load path follows the wheel into the stainless steel axle. The axle transmits the landing loads only to the mounting brackets; there is a clearance between the bronze bushing and the main strut. The mounting brackets then pass on the loads to the main strut through four AN-4 bolts. The wheel is tightened with two lock nuts, and the bushings of the wheel are only contacted with o-rings. The aluminum spacer prevents the bolts of the wheel from coming into contact with the bolts of the mounting bracket. More details on this assembly are given on DR 107-13.

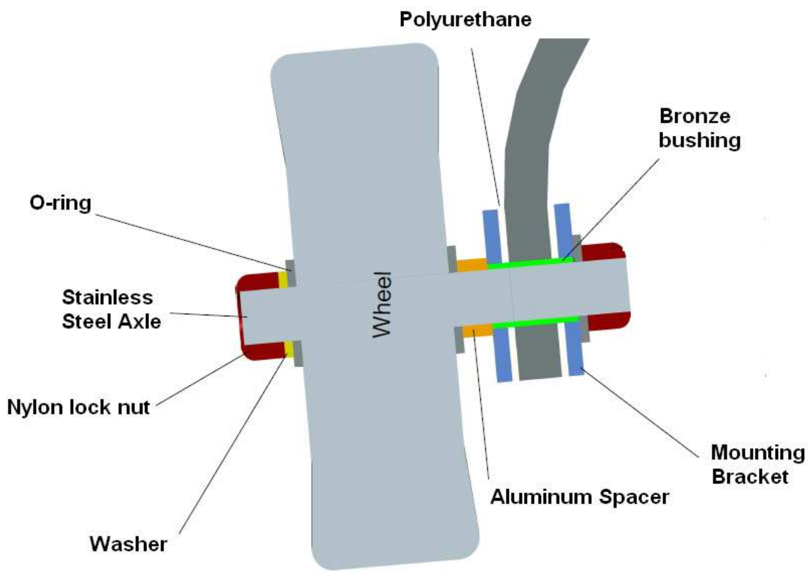


Figure D-101: Components in the wheel attachment

6.3.4 Failure of the Main Strut

During assembly of the aircraft on March 12, 2009 the main strut failed through delamination and it rendered the main strut unusable, this lead to a complete redesign of the main landing gear. From the requirements described in DR 87-10, the main strut should safely support 175 lb in a static position. However, the main strut started to flex gradually when the fuselage and wings where assembled (approximately 80 lb) and delamination was visible after the engine installation (approximately 100 lb). This occurrence is discussed in further detail in DR 107-13.

6.3.5 Weight and Recommendations

The main strut and wheel assembly have a combined weight of 12.6lb; the breakdown is shown in Figure D-102. Even though the weight of the main landing gear is under the 14lb objective, some more weight reductions can be implemented. A change of material from stainless steel to aluminum should be considered and analyzed for the main axle and mounting brackets. Also, if the budget allows, purchase of new lighter wheels with a larger tire diameter.

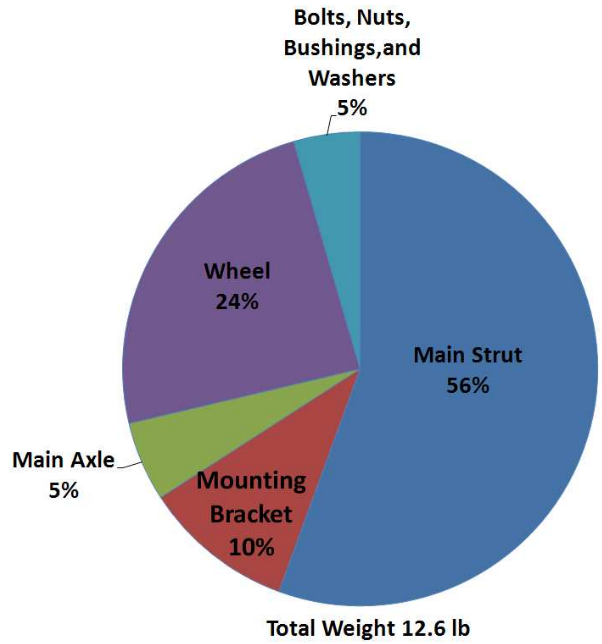


Figure D-102: Weight distribution of the main landing gear

Regarding the failure of the main strut, a redesign of the composite structure should be done. Also the ABAQUS finite element analysis performed on this new design should be verified through sample testing. Since the delamination occurred at the shoulders of the strut, testing should be done to verify the accuracy of the ABAQUS analysis; the results should be taken into consideration for the new design. The tests should include both static and dynamic loads. For further details, refer to DR 107-13.

6.4 Aluminum Main Landing Gear

After the failure of the composite main landing gear, an aluminum main landing gear concept was chosen as the best choice for the Geosurv II prototype. Due to the complex geometry of the landing gear (required because the centre of gravity is aft of the attachment bracket), the time constraints, the relatively easy machinability, and the team's familiarity with aluminum: this material was selected. DR 107-14 explains the details of this design.

6.5 Main Landing Gear/Fuselage Attachment Bracket

The attachment bracket fits directly under the carrythrough spar but it is attached with four bolts to the fuselage wall above the wing fairing. The main landing gear is connected to the attachment bracket with four bolts that allow for easy and fast assembly and disassembly. For the detailed design of the attachment bracket refer to DR 97-19 and for its manufacturing process DR 107-12. Figure D-103 shows the location of the attachment bracket in relation to the fuselage and main landing gear.

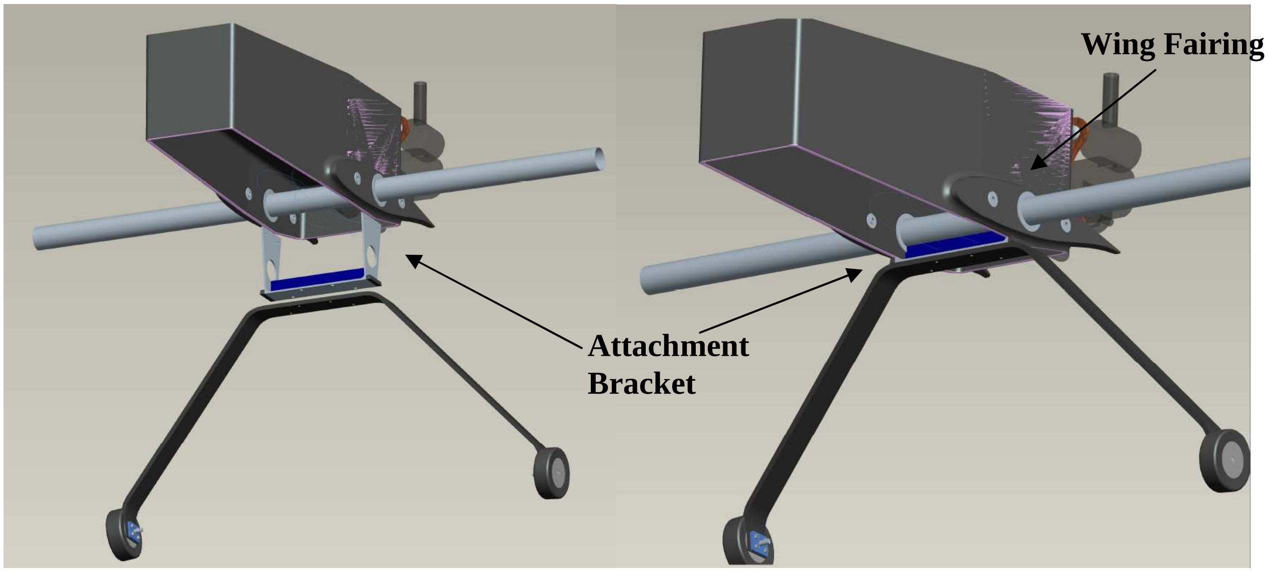


Figure D-103: Location of the attachment bracket with respect to the fuselage

6.5.1 Requirements

The purpose of the attachment bracket is to transmit the landing loads from the main landing gear into the fuselage walls. The detailed requirements for the attachment bracket are described in DR 97-19. These requirements are based mainly on two factors: interfacing with the fuselage, and landing loads. Other factors include weight, and serviceability.

6.5.2 Design

The design shown in Figure D-104 is composed of three main components: the shear support, panel, and web. The shear support and panel has a ply configuration of $[(0/90)_6/(\pm45)_4/(0/90)_6]$ and the web is a foam core piece wrapped in 8 plies of carbon fibre. Figure D-104 shows the components of the attachment bracket with its overall dimensions. The attachment bracket is connected to the fuselage with two bolts on each wall. The carrythrough spar hole is meant to have a significant clearance to avoid any loads being transferred to the carrythrough spar.

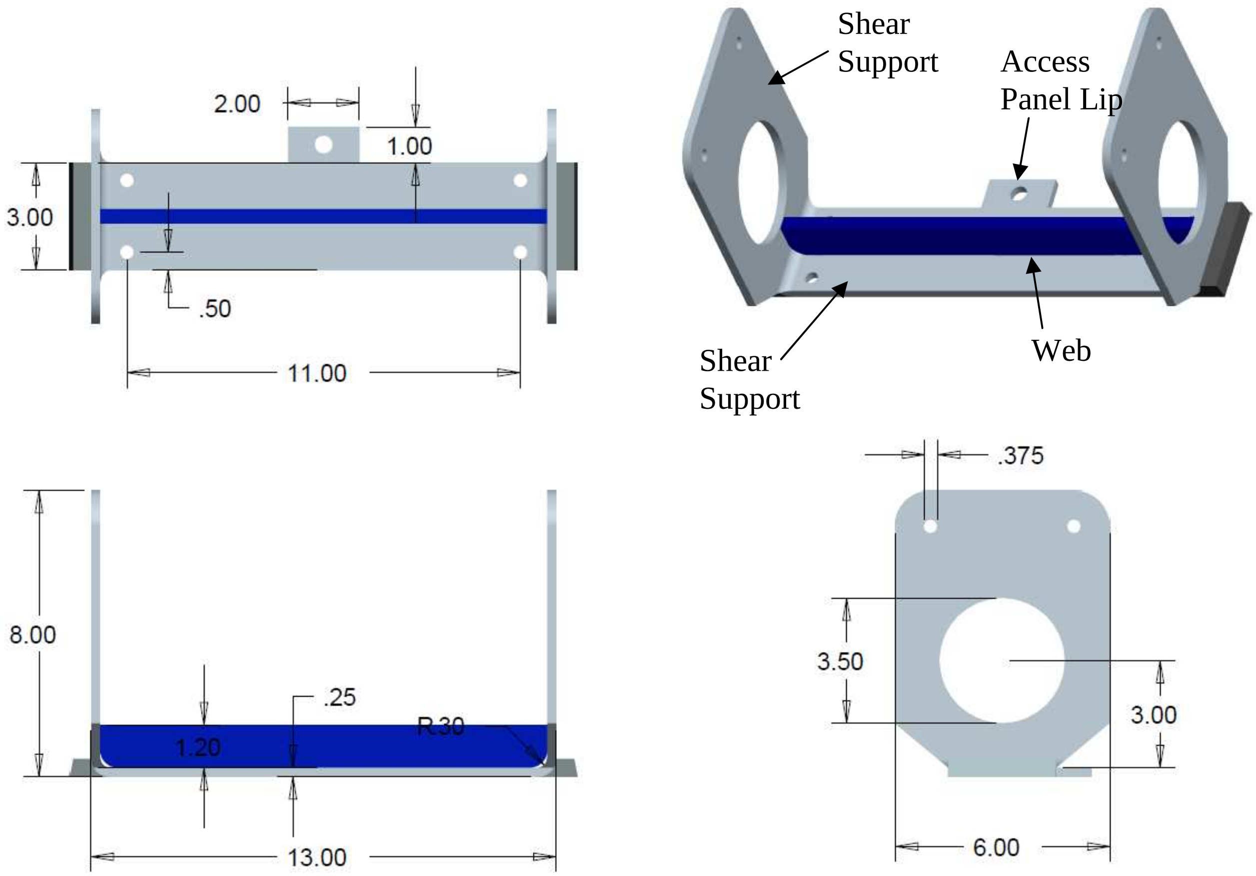


Figure D-104: Dimensions of the attachment bracket

6.5.3 Manufacturing

The manufacturing process for the attachment bracket is detailed in DR 107-12. This attachment bracket was manufactured in two components: the panel and shear support as one, and the web. The panel and shear support was infused in an MDF mould. A female mould was manufactured using a male mould. The male mould was made to the shape of the fuselage walls.

After the infusion, the panel and web were trimmed. After, the two components where secondarily bonded using Hysol 9430 and the carrythrough spar hole was cut, the attachment bracket was ready for installation. Figure D-105 shows the attachment bracket at this stage.

The attachment bracket and main landing gear holes were drilled simultaneously. Once the main landing gear and attachment bracket bolted together, this complete assembly was rigged into the fuselage and the carrythrough spar holes were cut.

Figure D-105: Manufactured attachment bracket

6.5.4 Weight and Recommendations

A complete ABAQUS analysis should be performed to determine the optimum number of plies necessary to balance weight and strength. This analysis will also confirm the stress calculations in DR 97-19.

The weight of the attachment bracket currently stands at 2.26 lb. Figure D-106 shows a breakdown of the weight. Possible weight reductions are described in DR 97-19 and DR 107-12.

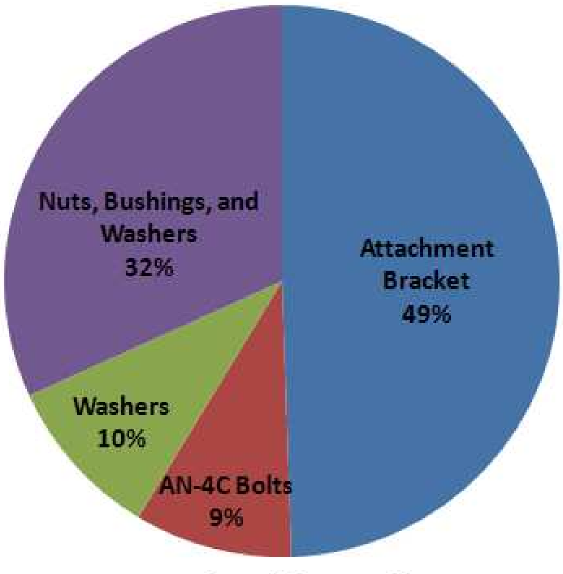


Figure D-106: Weight distribution of the attachment bracket

7. AIRCRAFT RIGGING

Aircraft rigging allows an aircraft to fly safely. Rigging begins in the construction of the aircraft. It allows for maximum performance by changing the interaction of multiple components. The GeoSurv II prototype aircraft consists of individually manufactured components. The fuselage is the primary component, the component to which all others are attached. The fuselage must be prepared first to ensure no problems will arise when installing the remaining components. Below is a list of the main components of the aircraft.

- Air Data Boom
- Avionics
- Control Surfaces
- Empennage
- Engine
- Fuselage
- Landing Gear
- Wing

The rigging procedure exists in two stages: mock assembly and final assembly. Signatures are required by both a supervisor and a witness during each step of the rigging procedure. The signatures verify that the procedure was carried out, and that each step was followed in the procedure.

7.1 Mock Assembly

The mock assembly determines the interfaces between two components. It verifies that the components line up, and determined the locations of holes for the fasteners, and the placement of equipment within each component. Modifications to interface designs are performed in this step to ensure that the components are properly aligned and can be assembled prior to painting. For a detailed step-by-step procedure, see DR 107-06.

7.2 Final Assembly

The aircraft can be completely assembled once each component has been painted. The holes of interfaces are to be re-drilled to remove any paint that may have accumulated in the holes. This will prevent the components and the paint scheme from being damaged. For a detailed step-by-step procedure, see DR 107-06.

8. PAINT

8.1 Paint Requirements

During flight, aircraft build up electrostatic charges on the external surfaces. The static charge presents a danger to human safety. If the aircraft does not dissipate the static charges during flight, it can result in a severe shock to anyone whom touches the aircraft once it has landed. Composite materials do not have the capability to act as a ground, as traditional metal aircraft do. Typically, composite aircraft must be grounded with the use of conductive paint and aluminum mesh. The paint distributes the electrostatic charge across the aircraft to the static wicks. The electrostatic charges are then dissipated during flight. It was determined that the prototype flight plan will not require conductive paint, as the aircraft will only be flown in clear conditions.

The aircraft paint must prevent any equipment on the aircraft, as well as the airframe itself, from overheating during flight. Since the aircraft is made from composite material, the aircraft must be white on the top surfaces since the aircraft is flying closer to the Sun, and white reflects sunlight. By reflecting sunlight, the composite structure will not absorb additional heat, which can compromise the structural integrity of the aircraft. The paint scheme must be easily seen, so that the operator of the aircraft will be able to track the flight of the aircraft, and in the event of a crash find the aircraft. The colour that is most noticeable in the daytime sky is yellow. The paint scheme of the aircraft is shown in Figure D-107 below.

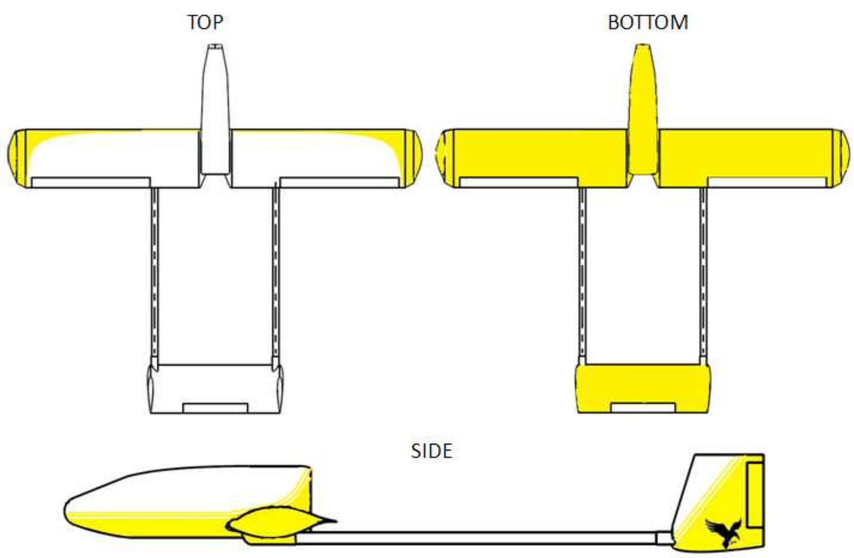


Figure D-107: Paint Scheme

8.2 Paint Products

A detailed description of the paint products, filler, and primer can be found in DR 107-05.

8.3 Paint Procedure

A detailed description of the painting procedure can be found in DR 107-05.

9. CONCLUSIONS

The GeoSurv II prototype structure is on the verge of completion, following the 2008/2009 academic year. Significant detail design was completed, primarily that of the interfaces between the main structural sub-systems. The VARTM composite manufacturing process was further developed and resulted in excellent quality components. The full structure was successfully assembled and the airframe finishing and painting is in progress. Initial structural testing was also executed and is planned to continue prior to first flights. Overall, continued efforts to reduce weight and improve manufacturing are being considered.

The fuselage access panel were redesigned and manufactured. Analysis of the rear bulkhead determined the need for additional reinforcement which lead to a redesign of the engine interface. A conservative analysis of the carry through spar determined that 8 plies are required for maximum manoeuvrability. The fuselage-wing interface was redesigned and implemented by the two subgroups. Finally, the fuselage avionic mounting method was determined and employed.

The wing interfaces were completed and the complete structures successfully manufactured and assembled. The wing tip fairings are the final element and will be completed in the spring. The wings were tested and determined to be adequate for initial flight, however are limited to a load factor of 4.2.

The manufacturing of the empennage structure was completed, the tailbooms purchased, and the empennage to boom interface components were successfully installed. A static load test was conducted. Although the tailboom stiffness requirement was not met, the tailbooms were found to be capable of carrying the predicted ultimate bending load.

Final design of the control surfaces was achieved, followed by manufacturing, installation and testing. Issues including flaperon interference, servo programmability, and weight are still being investigated and may require design changes.

The nose landing gear was successfully re-designed, manufactured, assembled and tested. The composite main landing gear design was pursued however led to delamination during the final assembly. A main landing gear attachment bracket was developed, and it remains functional. A new aluminum main landing gear design is in progress.

Finally, a rigging procedure was developed and is currently being executed throughout the complete airframe assembly.

10. RECOMMENDATIONS

The following are recommendations made by each sub-group of the 2008/2009 Structures Team for the incoming Structures Team in September 2009.

10.1 Fuselage

- Develop a repeatable process that can be used to ensure alignment and location of holes on multiple fuselages

- Agree with SGL on an avionics mounting method that could be included within the mission avionics bay (the rack that is currently installed is in response to a request made by the avionics team)

- Investigate methods to reduce the weight of the lower access panels

- Attempt to manufacture a carry through spar in house

10.2 Landing Gear

- For future teams, investigation into carbon fibre manufacturing techniques should be performed to create a nose gear that is potentially lighter.

- Polyurethane casting compounds should be investigated to produce compression disks that have higher dimensional accuracy.

- Perform structural tests on the composite main strut to investigate the reasons for the delamination.

- Explore alternative locations for the main landing gear/fuselage interface. This may eliminate the sweep back of the main strut.

- Future teams should look into ways of lightening various components in the design.

10.3 Wing

- Re-manufacture wing tip fairings in two parts that overlaps from male moulds

- Perform further testing, which includes:

- Bolt pull through test: determine the load at which an AN5 bolt head will pull through the wing skin

- Bearing test: determine if the lay-up of the root ribs is sufficient in term of bearing strength at the shear bolt locations

- Perform all stages of wing assembly in assembly jigs (i.e. lower wing skin mould)

- In future, consider redesign and manufacture of root ribs which are capable of withstanding ultimate loads at the shear bolt locations if necessary

- Investigate the feasibility of alternative matrix and reinforcement materials for weight reduction

10.4 Empennage

- Modify the design of the foam cores to incorporate the control surface cut-outs

- Investigate the closed cavity bag moulding technique for manufacturing future empennage components - this will allow the entire structure by infused at once while decreasing labour and improving the surface finish

- Optimize the foam core structure by switching to a structural grade foam and increasing the size of the lightening holes – thus reducing weight

- Optimize the lay-up of the tailbooms to incorporate more 0° plies (consider uni-direction fabric) and also possible drop-off plies

- Based on the results of flight testing, modify the design of the empennage structure to account for required adjustments to angle of attack or control surface sizing

- Consider an inverted V-tail design in order to significantly reduce the weight of the empennage, as well as reduce the effects of propeller turbulence on the empennage.

10.5 Flight Controls

- Design a fairing to encase the empennage servos and linkages to prevent damage and reduce aerodynamic drag. This fairing can be made out of ABS plastic using vacuum forming equipment

- Consider installing inserts or helicoils in the SPF inserts in the empennage trailing edge and control surface leading edges (there is a concern that the threads created in the wood will become loose and break after repeated fastening and unfastening of the screws)

- Perform fatigue testing on the plastic hinge to ensure that repeated bending under load will not crack or damage the plastic

- Develop and maintain a pro-active inspection technique for checking the interfaces and control systems before each flight

- Perform function and load tests on the control surfaces, measuring the power required to deflect the control surfaces and withstand loads

- Investigate alternative manufacturing methods for the flaperons that will reduce their weight

- Redesign the flaperon actuation system to achieve the required maximum deflection angle without interference

- Investigate the programmability of the existing servos to minimize current drain

10.6 Analysis

- Perform a verification analysis comparing ABAQUS analysis results to experimental results

- Perform an FE analysis on the current rear bulkhead design to determine the effect of the plywood reinforcement

- Perform an experimental test on the carry through spar

- Perform an FE analysis to determine the effects of the carry through spar on the fuselage side walls

- Perform an FE analysis on the new main landing gear design

10.7 Other

- Further investigate the benefits of using conductive paint

- Perform experiments on paint conductivity

- Compare various paints from different suppliers

- Investigate the use of aluminum mesh embedded in the composite structure to dissipate electrostatic energy

11. ACKNOWLEDGEMENTS

The Structures Group would like to thank:

- Prof. Straznicky and Prof. Laliberte for their support and guidance throughout the academic year
- All the lead engineers for their participation in the UAV project
- Graduate students Mario Mahendran and Shashank Pant for their assistance with manufacturing and analysis
- SGL for their continued support of the UAV project
- Hexcel Corporation for their generous donation of carbon fibre
- MAAE Machine Shop staff for their support and patience

12. DESIGN REPORTS

DR 77-01: Inverted V-Tail Empennage

DR 77-08: Spar Sizing

DR 83-02: Updated Load Analysis and Methodology

DR 87-05: Fuselage Finite Element Analysis

DR 87-41: Empennage Prototype Construction

DR 97-07: Manufacturing Tests for the Main Landing Gear

DR 97-04: Rear Bulkhead Finite Element Analysis

DR 97-16: Wing-Boom Interface Design

DR 97-11: Wing Support list

DR 97-18: Carry Through Spar Analysis

DR 97-19: Main Landing Gear/Fuselage Attachment Bracket

DR 107-03: Wing Tip Fairing Design

DR 107-05: Aircraft Paint

DR 107-06: Rigging Procedure

DR 107-07: Wing-Fuselage Interface

DR 107-12: Manufacturing of the Attachment Bracket

DR 107-13: Manufacturing of the Main Landing Gear

DR 107-14: Second Main Landing Gear Design

DR 107-15: Wing Assembly

DR 107-16: Wing Bearing Test

DR 107-17: Wing Structural Test

DR 107-18: Wing Weight Breakdown

DR 107-19: Bonding of Wing Components

DR 107-20: Empennage to Tailboom Interface

DR 107-21: Tail boom Design and Testing

DR 107-22: Wing Access Panel Design

DR 107-23: Wing Fuselage and Boom Interface Installation

DR 107-24: Wing Cutting Methods

DR 107-25: Wing-Boom Interface Testing

DR 107-28: Manufacture of Empennage Control Surfaces

DR 107-29: Completion of Empennage Manufacturing

DR 107-30: Empennage Servo Mounting