The

E-μ

Presented to:

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Table Of Contents:

Introduction

The goal this semester was to design an easy-build sailplane. While the sailplane had to fulfill many constraints it was decided to put more emphasis on simple fabrication rather than focus on performance. This Sailplane Design and Fabrication class has been going on for twelve years, and in that time, the class has had many accomplishments, but as yet, no full scale project has been completed. This fact drove the requirement that this design be easy-build. This main constraint was considered while designing the E-μ. The design group was split into subgroups, each completing analysis on individual components of the design. The groups met routinely to collaborate upon the overall design of the aircraft and ensure the individual work fit together correctly. This new research, along with knowledge of past projects, has guided the design of the E-μ.

Design Requirements

The general focus of this project was to design a plane by the end of the semester that would be able to be built by our class in two to four semesters and pass relaxed JAR requirements. The fabrication process is to use materials available in the Sailplane laboratory and include modular design, where possible, to keep the E-μ relatively inexpensive. The performance requirements specified that the aircraft have a minimum L/D_{max} of eighteen, minimum velocity above twenty knots and maximum velocity of sixty knots. For crashworthiness the glider must survive twice the minimum sink rate at impact. Using these requirements as guidelines, the group proceeded with the design.

Design Specifications

Fuselage

The fuselage was designed to allow the proper height for strut attachment and room for pilot comfort. The struts converge forty-three inches below the lower surface of the wing. This distance was used as the maximum fuselage height. A solid bulkhead would be required for structural purposes, so the seat must be placed ahead of the strut attachment. The struts attach to the wing at the spar, which is located at 40.5% of the mean chord. If the seat were placed directly in front of this bulkhead then the pilot would have a restricted vertical view due to the leading edge of the wing. To avoid this situation, the seat back was placed at the leading edge of the wing, Fig 1. The cockpit was laid out around the dimensions of “average man,” which were acquired from the “tumbleforms” web site.

Cockpit

The cockpit layout was designed around the accessibility of all controls while still maintaining the freedom of movement of both the pilot and the controls. The seatback is movable to allow for access to the battery and other stored items. It will rest against supports attached to the bulkhead ahead of the strut attachment. The fuselage bulkheads will be solid except for those located in the cockpit, which will be one inch thick semihoops. Lap belts will be attached to the bulkhead supporting the seat, while shoulder belts will be attached to the bulkhead aft of the seat, Fig. 2.

Wheel brake and control stick pulley lines will run through tubes underneath and around the seat. The wheel brake will be located just in front of the seat, underneath the pilots left leg. The control stick and pulley-housing unit will be between the pilot’s knees. Dive brake will be on the left side of the cockpit and will extend eighteen inches from a point just in front of the pilot’s shoulder, forward, Fig 2.

The rudder pedals will be on an adjustable bolt, driven by a crankshaft between the pilot’s legs. The rudders will have a three inch forward and aft range of motion. Rudder cables will be piped under the seat in the same manner as the control stick and wheel brake lines.

The constant pressure reservoir will be located forward of the first bulkhead. The total energy and pitot-static tubes, located on the nose of the aircraft, will feed into the control panel along with the constant pressure reservoir. The control panel will contain an airspeed indicator, an altimeter, a variometer, a compass, tow hook release and a radio to yell “MAIDEZ” on as one careens towards mother earth. The design also includes a large air vent located directly above and in front of the pilot’s head. Conversely, if a canopy is added to the design, a secondary air vent directly above the control panel is also planned. A canopy may be added if time allows or as a post-construction improvement.

Wing Fabrication:

When designing the wing, ease of construction and price were taken into consideration. First the spar design was started. A carbon tube spar, an aluminum spar and the Sparbon (short for the carbon spar no longer named the Marske spar) were all considered. The carbon tube spar will not be used because of the high price of the carbon tube. The aluminum spar was also considered and would work but the weight would be near 70 pounds for just the spar. The Sparbon was decided as the best choice because of the complete weight of only 40 pounds, it would be easy to build, and the price will be under $300.

First we will buy prefabricated Laminated Panels from Aerospace Composite Products. After printing out the SM-701 airfoil and tracing it on the Laminated Panels, the rib may be cut out.

The D-tube is constructed using the front of the laminated rib panels. Balsa is then laid over the ribs and covered with glass, Fig. 4. The spar uses Graphlite Carbon Fiber Rods (http://www.aviasport.net) for the main structural strength. Construction of the spar is as follows; a mold is built out of 3 lengths of wood, the centerpiece is tapered and then the two other pieces of wood are glued at 90 degrees. After waxing the mold, one layer of glass is laid in the mold. The first rod is then laid in the top and bottom corners, epoxy will be spread over the rod; another shorter rod will be laid on top of the first rod. This process will continue until enough rods have been placed on top of each other for the appropriate strength, 9 per cap for the E-μ. Another layer of glass is then placed over the rods. A sheet of foam is laid in between the rods and then covered with 2 layers of glass laid over the foam and rods. The whole spar then will be vacuumed bagged.

The D-tube and the spar are joined together. The ribs are then glued to the spar at the proper spacing, Fig. 4a. A trailing edge plywood piece is then glued to the wing. The push pull wires are installed, the spoiler box and then the aileron mounting pieces are glued in their proper places. The wing is now covered with fabric and the fabric is doped.

Horizontal and Vertical stabilizers:

The horizontal and vertical stabilizers will be constructed much like the wing, with only minor differences. The structure will consist of ribs and stringers, covered by dope and fabric. A D-tube is not needed as long as the ribs are spaced closer together. For simplicity of manufacture, and with only minor performance penalties, a conventional tail design will be used as opposed to a T-tail or V-tail alternatives

Controls

The controls will be cable driven. A cable will run from the stick and two rudder peddles via tubing to a “mixer-box” behind the seat. This will not be a conventional mixer-box with bell cranks and push/pull rods; rather it will consist of a system of pulleys to direct the proper cable to the proper control surface. From this mixer-box, cables will run to port and starboard ailerons, elevator and rudder. Since this aircraft will not be able to achieve relatively high velocities (high being over 150kts) the cables will not be subjected to forces that could cause them to stretch.

Pulleys will replace bell-cranks to redirect the cable where needed. For each aileron, a pulley will be located at the wing fuselage union, and at the aileron driver. The rudder and elevator will require similar redirection. In addition several support wheels may be added to ensure the cables do not sag.

Landing Gear

The landing gear consists of a forward skid and a wheel located behind the fuselage, Fig 5. At this time the wheel is not retractable for ease of construction; however if time allows, a retractable wheel would reduce the drag considerably.

Weight Analysis

Estimating the weight of the individual components, and summing them created the weight buildup. Breaking the glider up into more basic components allowed for more accurate estimates to be made, reducing the error of the total weight estimate. When possible, such as with the skin and bulkheads, the component weights were found using a lb/ft^2 given value, and an estimate of the total area of the material needed. Otherwise, estimates for the components were made using the best possible, realistic guesses while still ensuring a slight overestimate, for a safety margin. An addition, overall margin of error of ~5% was added for safety, and to round out the weight estimate. The final weight of the glider was 265.5 pounds empty, 440 pounds full, Fig 6.

Wing Structural Analysis

The wing structural analysis involves many different elements and is integral to the design of the aircraft. The spar carries the shear and bending moments. The d-tube carries the shear due to the aerodynamic moment. The ribs reduce buckling and transfer aerodynamic forces from the skin to the spar and d-tube.

The first step in the analysis was to derive the V-n (Velocity-load factor) diagram using the JAR-22 requirements and guidelines for maneuvering loads and gust loads, Fig 7a & 7b. These requirements specify allowable load factors for different speeds. The maximum positive load factor was 5.5 and the maximum negative load factor was –3.5. These numbers were used for the bending, shear, and strut loads. The maximum design speed was 85.5 knots. This maximum speed was used for drag loads, maneuvering loads, and d-tube torsion loads.

The second step was to find an approximate lift distribution. This was done by finding a simple function for a line that closely resembles the actual lift distribution. The next step was to determine the spar loads of which the most significant was the bending moment. The strut attaches to the wing at one-third half span, or 8 1/3 ft from center line. The presence of the strut greatly reduces the bending moment on the spar. An exact solution was found for a beam in this loading case. The beam is pinned at the fuselage, pinned at the strut, and free at the tip. Therefore there is no displacement at the fuselage and strut junction and no shear or moment at the tip. These boundary conditions along with the lift distribution function allow for the beam bending solution for the spar. The solution shows displacement, slope, moment distribution, shear distribution, and lift distribution of the spar. The maximum bending moment was derived from this graph, Fig 8. It was found to be 19000 ft-lbs at the strut with a load factor of 8.25. This load factor is found from the calculated load factor of 5.5 times the factor of safety of 1.5. Since carbon fails near its yield point, the factor of safety was used at this point in analysis.

The spar consists of carbon rods as the spar caps with four +/- 45 degree layers of 7781 fiberglass holding them together. The carbon rods react the compression and tension at the caps, which react the bending moment, while the layers of fiberglass react the shear and the compressive force between the caps. The spar location along the chord is .381c (16 inches from leading edge at the root). This point is about three inches in front of the maximum thickness, though the thickness at this point is less than 5 percent less than the maximum thickness. This location supports a moment of inertia nearly equal to the maximum moment of inertia of the spar while minimizing buckling of the skin by making the d-tube smaller. The following equation was used to find the required moment of inertia of the spar:

$$I=(M*Y)/\sigma \text{ - Bending moment equation}$$

Here, Y is the height from the center of the spar to the caps at the maximum bending moment location, M is the maximum bending moment from the beam solution, and σ is the yield stress of the material. The maximum height of the spar at the strut is .4475 ft and the compressive yield stress of the carbon rods was 275ksi. The tensile yield stress was 320ksi, but 275ksi was used to simplify analysis and overdesign the spar caps to ensure safety. This solution yielded a required moment of inertia of 2.2261 in^4 which means a total of 16.59 rods of size .220 x .092in would be required. This number was rounded to 18, 9 rods on the top and bottom caps, so that the caps were of equal size.

Next, the loads in the web were calculated. From the beam solution a maximum shear force was found to be 1291 lbs. The maximum shear was at the strut where the height is 5.4in. Using the equation:

$$Pxy=V/d$$

where, V is maximum shear and d is spar height, the shear load on the web was calculated to be 240 lbs/in. The allowable load was found by multiplying the yield stress of the web by the web thickness. The yield stress of 7781 fiberglass is 45ksi and the thickness is .08in per ply, which for the E-μ design is multiplied by four. The allowable load is 1440lb/in, which is much greater than the calculated load on the web. The compressive force on the web due to the bending moment is found using the equation:

$$Py=(2*M*b/d\text{^}2)*(\sigma_{maxcap}/E_{cap})$$

where b is the cap width, d is the spar height and M is the maximum bending moment. From this equation the maximum compressive force is 77.2lb/in. From the equation:

$$Py_{allowable}=\sigma_{plyweb}*t_{web}$$

the allowable compressive force for a four layer web is over 192lb/in. This assumes the fiberglass has a strength 13% of its maximum value in a 45 degree orientation. With factors of safety this large the web could be shrunken three layers, though buckling and de-lamination problems could arise. A three layer web with 18 carbon rods would weigh approximately 23 pounds while a four layer web with 18 carbon rods would weigh approximately 25 pounds. This is a calculated value so the actual weight will be between 25 and 30 lbs. These numbers are very conservative and include resin on a 50 percent content of 9oz. Fiberglass along with 18 carbon rods weighing 13lbs/ft along the full span of the wing and ¼in 4lb foam panel in the web. The root and strut fittings were not included.

Strut Structural Analysis

The strut needs to support a load of 3550 lbs. From the equation I=(Pcr*L^2)/(Pi^2*E) for a 6061 T-6 spar the required moment of inertia is .42032in^4. The length of the strut was 108in. From this calculation we will find an appropriate sized airfoil shaped strut.

D-tube Structural Analysis

The D-tube is located at the leading edge of the SM701. The primary job of the D-tube is to resist torsion due to lift, support the spar in resisting shear and resist drag forces. First shear flow was calculated along each panel length. This gives max shear along each panel of the D-tube. With this information we can calculate how thick the panels need to be to resist the shear force. The calculated stress due to shear will be an over approximation since most of the shear will be carried by the spar.

Since the D-tube will be constructed from fiberglass 7781 and balsa wood, orientation of the fiberglass plies is crucial to resisting max shear and torsion. Currently there will be 3 plies of glass; each ply has an orientation of +-45 deg and a thickness of .008 inches. Lift is assumed to act at the D-tube. Torsion is due to the lift moment. This causes shearing in between the laminates. Layering the plies to resist this force can be done using methods learned in Aerospace 302. We will calculate the extensional stiffness in x and y directions and the in-plane shear stiffness, A_{11}, A_{22}, and A_{66} respectively. Interpretation of the calculation will enable us to reach maximum stiffness by using different laminate layering patterns. Since this is an iterative process also due to time constraints and difficulty, we are still crunching the numbers for structural stability of the D-tube. However because it is iterative the integrity of the D-tube will be unquestionable because the results will be solid.

Wing-Fuselage Union

The wing and fuselage will be joined at four points.

1. The two struts will attach to the bottom of a bulkhead in the fuselage.

2. The spars will come together at the top of the fuselage. The port side spar will have a flange added to the root side, it will extend over the starboard spar. A hole will be drilled through the flange and the starboard spar. Upon assembly of the wing and fuselage a steel dowel will be inserted through the hole and a cotter pin attach to secure the pin in place.

3. Eyehooks will be added to the root ribs. Upon assembly a pin will be inserted through each eyehook into a flange that is attached to the bulkhead just behind the seat.

Drag Build-Up

To calculate the total drag of the aircraft, the drag of the fuselage, wing, struts, and empennage were analyzed and then summed.

Fuselage:

To calculate fuselage drag, the skin was divided into nine rectangles. The width of each rectangle was found using the equation for the circumference of an ellipse. Multiplying the width by the length of the panel gave the area. We assumed that the first four panels would have laminar flow and the final five would be turbulent. The tail boom was divided into four segments and evaluated in a similar manner with all panels having turbulent flow. The skin friction coefficient for each panel was calculated using the equations:

$$C_{f\ laminar} = 0.664/\sqrt{R_e}$$

$$C_{f\ turbulent} = 0.0576/R_e^{0.2}$$

These values were then entered into the equation:

$$D = 1/2\rho V^2 S_{panel} C_f$$

This calculation was performed for the range of flight velocities.

Wing:

The drag of the wing was calculated by dividing the wing into three sections per half span, for a total of six sections. The mean chord of each section was determined, which allowed for the Reynolds number of each section to be calculated. The required coefficient of lift was found using the lift equation. These two values were then used to find the coefficient of drag from the C_l vs. C_d plots for the SM701, Fig 9. Using these C_d values the drag was calculated using the equation:

$$D_{profile} = 1/2\rho V^2 S_{ref} C_d$$

This was repeated for the range of flight velocities for each wing section. The drag of each section was then added to find the profile drag of the wing. The induced drag was calculated using the equation:

$$D_{induced} = 1/2\rho V^2 S_{ref} C_{di}$$

The coefficient of induced drag was calculated using the equation:

$$C_{di} = C_l^2/\pi e AR$$

The induced drag was calculated for the range of flight velocities. The induced drag and profile drag at each velocity were then added together to find the total drag of the wing at each velocity.

Struts and Vertical Tail:

The struts were assumed to be symmetrical airfoils at zero angle of attack. By analyzing several symmetrical airfoils, the coefficient of drag at C_l=0 was determined to be C_d=0.008. This allowed for a similar calculation of drag for the flight velocities using the drag equation.

The vertical tail airfoil will also be symmetrical. Using the mean chord of the vertical tail, the Reynolds number was determined. Using the Reynolds number and $C_l = 0$, the coefficient of drag was calculated from the C_l vs C_d graphs for the FX 71-L-150/20 airfoil. The drag of the vertical tail was then determined using the drag equation for the range of flight velocities.

Horizontal Tail:

The drag of the horizontal tail was determined using the same process as that used to find the wing drag. The lift coefficients for the horizontal tail were determined for trim flight. These values were used to find the coefficients of drag.

Total Drag:

Once the drag of the individual components was determined, the total drag of the aircraft was calculated. Total drag is equal to the sum of the drag of the components:

$$D=D_{wing} + D_{H\text{-}tail} + D_{V\text{-}tail} + D_{struts} + D_{fuselage}$$

This calculation was performed for the range of velocities. Once the total drag was found the drag coefficient at each flight velocity was determined using the equation:

$$C_D = \frac{W}{1/2 * r * v^2 * S}$$

Performance Analysis

The completion of the drag build-up allowed for the performance analysis to be obtained. The best L/D was calculated to be:

$$L/D_{max} = 21.19$$

This is above the required L/D_{max} of 18, Fig 10. L/D is calculated by using the drag coefficient and lift coefficient at each velocity:

$$L/D = \frac{C_L}{C_D}$$

The minimum sink rate was found using the equation:

$$V_{sink} = -\sqrt{\frac{2 * W}{r * S}} * \left(\frac{L}{D}\right)^{3/2}$$

This results in $V_{min\ sink}$ in ft/sec. The minimum sink rate is:

$$V_{min\ sink} = 136.12 \text{ ft/min @ 24kts (Fig. 11)}$$

A graph of C_D vs C_L^2 was plotted, Fig. 12. The slope of this graph, k, is used to determine the Oswald efficiency, e_o, and the y-intercept, which is the coefficient of zero lift drag, C_{Do}. The Oswald efficiency is equal to:

$$e_o = \frac{1}{k * p * AR}$$

These values are:

$$e_o = 0.67$$

$$C_{Do} = 0.0206$$

Crashworthiness

The design requirements specify that the aircraft be able to survive a twice min-sink rate hard landing. To fulfill this requirement the fuselage contains stringers on the bottom half. These stringers are made from an epoxy-glass sandwich. Foam is cut into a trapezoidal shape and epoxy-glass is laid-up over this shape. The stringers are then attached to the inside of the fuselage bulkheads. This design was first developed to strengthen the Griffin design and make it crashworthy and should be quite simple to include in the new easy-build design.

Time estimate

A long and in-depth calculation consisting of high order calculus, trigonometry and quantum physics was used to derive the time estimate. Since the method used in this calculation consists of very impressive words and terms, the arrived at estimate of eighteen months is highly probable.

Contained within this estimate are one semester allotted for wing construction, one semester allotted for fuselage construction, and one semester allotted for final assembly. Empennage construction can be completed simultaneously with the wing.

Cost

The cost buildup was created by adding the cost of all the materials to be purchased or used during the construction of the E-μ. Price per square foot for each type of component was obtained from Aircraft Spruce and Specialty, and other aircraft catalogs, and then multiplied by the estimated amount of material that will be used. An error margin of 10% was added to the cost, along with a factor of 6% to account for potential sales tax. Alcohol money for the post-flight party was not included in the cost buildup, but should not exceed $500, if funded by the department, or $100 if paid for out of pocket by the partygoers, via a kitty, or similar method of money collection. The total, final, adjusted cost for the glider was $3835, Fig 13.

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Appendix