**Aircraft Electrification Case Study**
**Project Final Report**

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# 1. Project objectives

With the growing challenges of climate change, electric and sustainable aviation has gained a lot of popularity and importance. Electric aircraft would offer a more sustainable solution, reducing greenhouse gas emissions, and would eliminate costs linked to fuel consumption, which is a large portion of the operating costs of an aircraft. While a lot of aircraft are fuel efficient, for environmental and cost reasons, this can only be achieved to a certain extent. This is where electric aircrafts can come in and solve these issues. However, as with any new system, electric aircrafts have limitations that need to be resolved. A major limitation is the range of operation of the aircraft. Since batteries are required to power the electric motor, a large mass is added to the aircraft due to their low power density compared to fuel, which leads to a reduction of the payload capacity and the overall effective range of the aircraft. For this reason, hybrid aircraft have been developed to combine the benefits of both a gas turbine engine and an electric motor. However, for the purpose of this project, only pure electric aircrafts will be considered.

The goal of this project is to evaluate the electrification of an aircraft, specifically the AA-1 Yankee Clipper. This entails modifying this aircraft to fit an electric motor from MGM COMPRO, adding the batteries required to power the motor and the controllers that regulate the power of the motor. This will require the design and analysis of an engine mount to fit this motor, which must adhere to FAR regulations, and a material analysis. Once the motor is chosen and implemented, the changes to the flight must be analyzed with a performance analysis. Since this new motor will require new avionics components, the existing system will be modified to implement all necessary components.

# 2. Configuration

*Power Considerations for Motor*

The American Aviation AA-1 Yankee Clipper was designed in 1962 by Jim Bede and was certified for flight in 1967. Only a small number of units were produced in the early seventies, but it was a lightweight and relatively low-cost small 2-seater aircraft. It is powered by a single Lycoming O-235-C2C 108hp, 2600 rpm engine and has a dry weight of 982 lbs [1]. The dry weight doesn't include any payload, passengers, or usable fuel. The speed provided is the maximum speed of the engine during take-off.

To be able to extract enough power from the electric aircraft counterpart, the electric motor must match the main performance parameters, such as power and RPM, while minimizing weight and size due to the limited space available. To reduce the search range of the motors, only motors from MGM COMPRO are considered in this project.

By converting the horsepower to power, the maximum power that the electric motor needs to supply can be found. Here, 108 horsepower corresponds to approximately 80 kW, so all motors with a peak power less than 80 kW are automatically eliminated from the list of potential options. The only motors left are the following:

- REB 90, which has a peak power of up to 80 kW, with the continuous power being around 60-70 kW, a maximum torque of 300 Nm, and a speed ranging from 1500 to 4000 RPM. It has a weight of 23.8 kg and a diameter of 27.02 cm. It needs to be powered with 400-800 V. [2]
- MGM BLDCin 200 kW, which has a peak power of 200 kW, with a continuous power output of 150 kW, a peak torque of 370 Nm, a continuous torque of 240 Nm, and a maximum speed of 6500 RPM. It has a weight of 53.5 kg and a diameter of 28.32 cm. It needs to be powered with 800 V. [3]
- MGM BLDCin 400 kW, which has a peak power of 400 kW, with a continuous power of 350 kW, a peak torque of 540 Nm, and a maximum speed of 8000 RPM. It has a weight of 129.7 kg and a diameter of 33 cm. It needs to be powered by 800 V. [4]

By looking at these values, the last option can be eliminated since it is a lot more powerful than the Lycoming engine. If the aircraft's weight was changing by a lot, this would be an interesting option, but since it isn't, it is too extreme of an option.

*Torque Considerations for Motor*

Torque should also be considered. The Lycoming engine has a torque of approximately 218 lbs-ft, which corresponds to about 296 Nm of torque. This was calculated using the given speed in RPM and the horsepower of the engine. Both the REB 90 and the MGM BLDCin 200 kW would match this torque, so they are both valid options from this parameter point of view.

*Weight Considerations*

Another important parameter to look at is weight. While the electric motor is much lighter than the gas turbine engine, the added weight for the batteries and controllers need to be considered. The Lycoming engine has a dry weight of 245 lbs, or 111 kg [5]. Thus, the modified aircraft without the engine will weigh 982-245= 737 lbs. However, the aircraft can also drop weight by getting rid of the fuel system and ignition items; the total weight removed is estimated (rounded numbers) below.

**Table 1: Total weight that can be removed from the empty weight of the fuel-powered AA1 Yankee Clipper [6, 7, 8]**

| | |
|---|---|
| Engine (Lycoming O-235, dry) | 245 lbs |
| Original engine mount | 15 lbs |
| Induction & Carburetion | 10 lbs |
| Fuel delivery & controls | 25 lbs |
| Ignition | 10 lbs |
| Exhaust system | 20 lbs |
| Oil system | 15 lbs |
| Engine baffles & ICE-only brackets | 5 lbs |
| TOTAL | 345 lbs |

Thus, the airframe removals can be conservatively approximated at 345 lbs, and the fuel and oil at 140 lbs [9], making the total weight removed equal to 485 lbs. This means that the added weight of the electric motor, controller, and batteries can be a maximum of 485 lbs in order to not exceed the maximum weight that the aircraft must be at take-off, which is 1500 lbs according to the Pilot Operating Handbook for the AA1A Yankee Clipper.

With this information, it was deduced that the 200-kW motor would be the best choice for this project as it provides more power than the gas turbine engine, has a similar torque, is smaller, and would weigh approximately the same with the added batteries.

*Batteries and Controllers*

Now that the motor is selected, batteries and controllers must be chosen.
The battery for the electric motor was selected to maximize power output while minimizing added weight. The Husky 2P50 offers low mass and high-power capability, with a specific energy of 215 Wh/kg. [10] Battery selection also considered system voltage, since the Husky 2P50 from Mobius Energy has a nominal voltage of 800 V. This battery is therefore ideal for its voltage, as well as weight, size, and power output.

A controller is also needed to control the motor and ensure that it operates safely. The two preferred controllers were:

- HBCi 400800 320 kW, peak power 400 kW, voltage 120-800 V, max continuous current 400A [11]
- HBCi 400400, 160 kW, peak power 200 kW, voltage 120-400 V, max continuous current 400A [12]

While both have a maximum efficiency of 99% and a weight of 4 kg or 8.82 lbs, the clear choice is the HBCi 400800 due to its compatibility with our 800V motor bus voltage.

With the chosen motor weighing roughly 120 lbs, and the controller 10 lbs, the remaining weight for batteries and additional electricals would be 485 – 130 = 355. The electricals can be approximated at 35 lbs, allowing for 320 lbs of battery. This means that 4 batteries weighing 75 lbs each can be fit, and 20 lbs are available for extras.

*Component Sizing*

The sizes of the motor and the other components are also important factors to consider. Their arrangement in the engine mount is crucial for the aircraft to fly with optimal conditions. According to the PDF provided, the engine mount is mounted on the firewall, which is a 36” by 24” frame [13]. As for the chosen components, the motor has an outer diameter of 12” and a length of 4.4”, the controller has a respective length, width, and height of 10.8”, 8.7” and 5.6”, and the batteries have dimensions of 29.5” in length, 26.7” in width and 4.9” in height [3, 11, 10]. This means the batteries will be the biggest architectural challenge as they are the biggest and heaviest components.

To fit the batteries inside the engine mount, its length is crucial. The batteries could be put flat on the ground if the length is big enough. Its height being only 4.9” creates a large enough gap for the engine to be above or beneath them with clearance. The same goes for the controller. Alternatively, batteries could be positioned in the wings where the fuel tanks were. This would replace the weight created by the tanks and free some space in the engine mount. All of this must be carefully thought of and designed to make the plane flyable and safe.

*Propeller*

The FAA AA1A type certificate document [14] offers 4 types of McCauley propellers that can be installed and used on this aircraft. The weight, rpm, and diameter for each of these propellers can be found in the table below.

**Table 2: RPM, Diameter and Weight for 4 Different McCauley Propellers**

| | Static RPM at max permissible throttle setting | Diameter | Weight |
|---|---|---|---|
| McCauley Model 1A105/SCM-7157 fixed pitch propeller | Max: 2300<br>Min: 2150 | Max: 71 in<br>Min: 69.5 in | 13.609 kg (30lbs) |
| McCauley Model 1A105/SCM-7153 and 1A105/SCM-7154 fixed pitch propellers. | Max: 2400<br>Min: 2250 | Max: 71 in<br>Min: 69.5 in | 30 lbs |
| McCauley Model 1A106/NCM-7157 fixed pitch propellers. | Max:2400<br>Min: 2300 | Max: 71 in<br>Min: 69.5 in | 23.5 lbs |
| McCauley Model 1A106/NCM-7153 hub and fixed pitch propellers | Max: 2475<br>Min: 2375 | Max: 71 in<br>Min: 69.5 in | 28.2 lbs |

We chose the McCauley Model 1A106/NCM-7157 fixed pitch propeller because of its minimal weight in comparison to the three other propellers.

*Weight and Stability Analysis of the Aircraft*

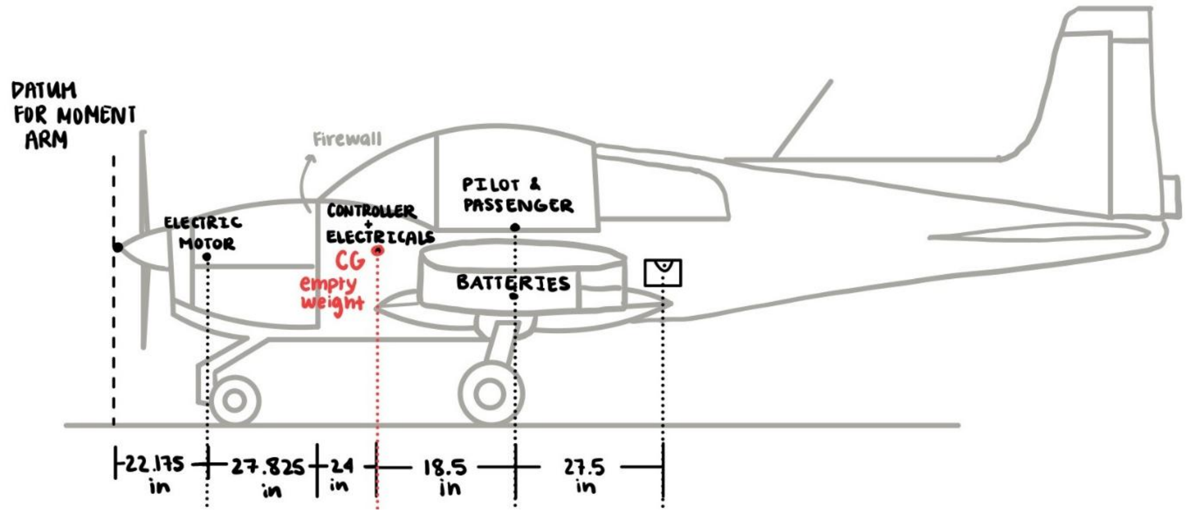


**Figure 1**. AA1A Electrified Yankee Clipper Schematic

**Table 3: Various Weights and their corresponding moment arms in the AA-1 Yankee Clipper**

| | Weight (lbs) | Moment Arm (in.) | Moment/1000in |
|---|---|---|---|
| Empty weight | 637 | 74 | 47.138 |
| Pilot and Passenger | 340 | 92.5 | 31.45 |
| Baggage | 10 | 120.0 | 1.2 |
| Electric Motor | 117.95 | 22.175 | 2.6155 |
| Battery | 288 | 92.5 | 26.64 |
| Controller | 8.82 | 62.75 | 0.553 |
| Additional Batteries and Electricals | 20 | 92.5 | 1.85 |
| **Total:** | 1421.77 | N/A | 111.447 |

All the moment arms used to compute the moment of these weights are measured from a datum located 50 inches in front of the firewall, which is a physical barrier that separates the engine from the cockpit [14].

The AA-1 Yankee Clipper uses the Lycoming 0-235-C2C fuel engine [1], which has a dry weight of 244 lbs. Our previously estimated weight for additional components within the fuel-powered AA-1 Yankee Clipper's empty weight, which is not required for the operation of the electrified AA-1 Yankee Clipper, was 345 lbs. The new empty weight of our electrified AA-1 Yankee Clipper can therefore be determined using the empty weight of the fuel-powered aircraft, 982 lbs [1], minus the unnecessary components (345lbs) to obtain a new empty weight of our electrified aircraft of 637 lbs. We considered that our electrified Yankee Clipper can carry a pilot and a passenger with additional baggage, and that the removal of a passenger and the baggage doesn't impede the stability of the aircraft and keeps us within the stability envelope shown in Figure 2 below. We assumed the positioning of our electrical motor to be in the location of the engine from the fuel-powered AA-1 Yankee Clipper, at the front of the engine mount, with its center of gravity located at the middle of the motor, estimated to be 28 inches before the firewall. The batteries for the electric motor are placed at the wings (where the fuel would have been), which begin at 84.5 inches from the datum, and are approximately 16 inches wide, therefore, the center of the wing is located relatively in line with the pilot and passenger at 92.5 inches from the datum [14]. Each battery weighs 72 lbs. The controller is located in the cockpit, slightly in front of the pilot and the passenger at 74 inches from the datum, which is where the airframe's center of gravity is, and is where any other additional batteries and electrical components that may be required to operate the controller, or any other systems within the aircraft are placed. The moment arms used for the pilot, passenger, and baggage were all obtained from the AA1A Yankee Clipper Pilots Operating Handbook.

In order to ensure that our electrified AA-1 Yankee Clipper satisfies all the weight and balance requirements and can remain stable during flight, Figure 2 below can be used.

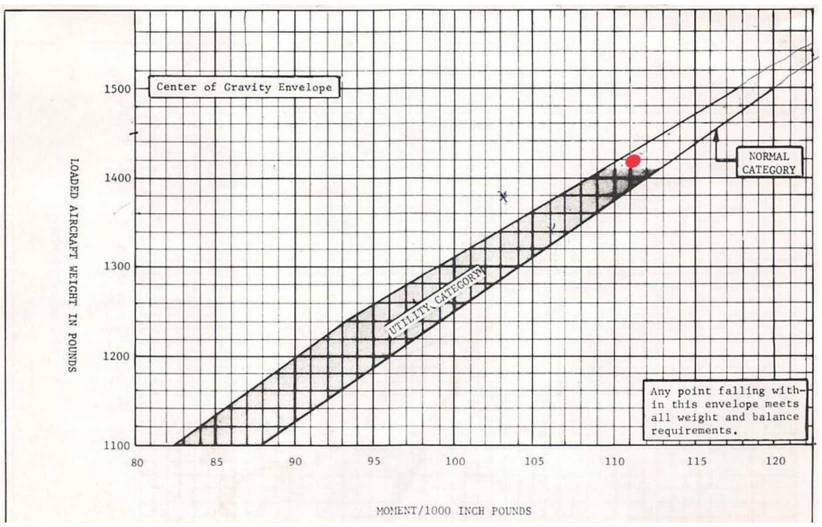


**Figure 2: Weight and Moment Distribution Graph for Stability of the AA1A Yankee Clipper (obtained from POH for the AA1A Yankee Clipper) [9]**

Our resulting aircraft weight and total moment calculated from Table 2 were 1421.77 pounds and 111.447 inch-pounds, respectively. By plotting this region in Figure 2 (the red dot), our electrified AA-1 Yankee Clipper lands within the envelope that establishes we have passed the weight and balance requirements for the stability of this aircraft.

## 3. Engine mount analysis

With the motor, batteries, and controllers chosen, the engine mount can now be designed and analyzed using the different load cases presented in the FAR 23 normal category aircraft to determine the maximum stresses that the part will experience. It will have to fit onto the front of the aircraft in the available space, attach to four connection points on the firewall, and ideally must not modify the original CG of the aircraft once the motor is installed to maintain stability [15].

*Positioning of the motor within the engine mount*

The motor needs to be positioned at the front of the engine mount to attach to the propeller, assuming the CG of the motor of length is at its center, the following diagram describes its placement in the engine mount:

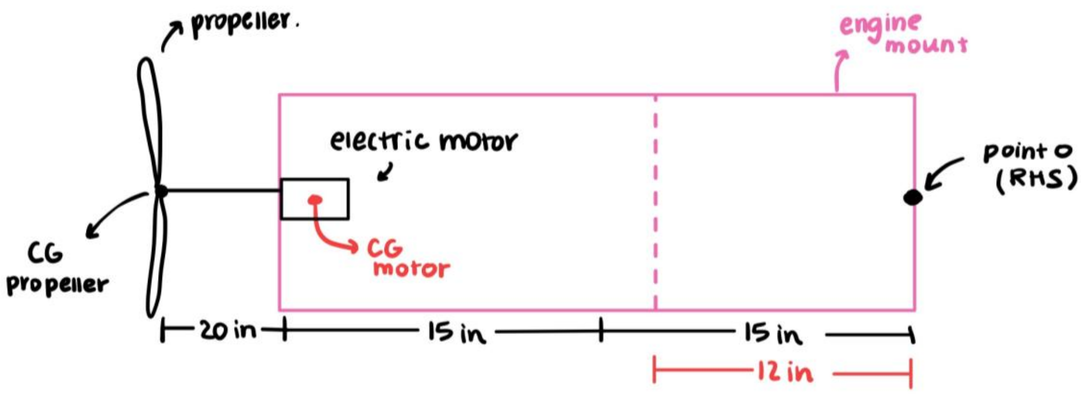


**Figure 3: Schematic of Engine Mount, Electric Motor and Propeller**

*Engine mount design and setup*

To not disrupt the CG location, the engine mount will fit in the already available space in front of the firewall of the AA1A Yankee aircraft. By superimposing the figures available in the AA1A POH, this distance can be estimated to approximately 31”. To ensure that the engine mount will fit properly, the total length chosen will be 30” for the engine mount. It is composed of the firewall, which separates the pilot cockpit from the engine and protects the crew in case of a fire [16], the middle frame, and the front frame. The firewall has 4 connection points, whose coordinates can be seen on Fig.4. The middle frame is set to a position of z=0 and is sized to be much wider than the motor, the size chosen is 10” by 10”. As for the front frame, the dimensions were chosen to match the attachment points of the motor. Since the technical drawings for the MGM BLDC 200 kW are not available, this location had to be estimated using the diameter of the motor, which is 11.15”. The attachment points are located farther away from the centre and were approximated to 6.5”.

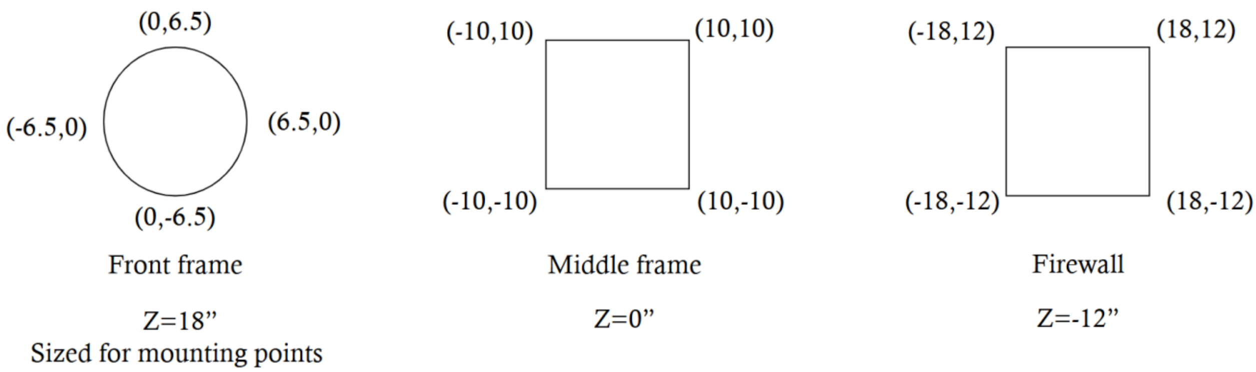


***Figure 4:*** Dimensions and sizing of the 3 engine mount frames

The length between the frames was chosen to allow for enough space for the motor between the front frame and the middle frame. The properties of the trusses that connect the frames can be found in Table 4
The full engine mount with trusses was modelled in Abaqus as a wireframe and can be seen in Fig. 5. Since the material needs to be both strong and lightweight, titanium is chosen, specifically Titanium Ti-6Al-4V (Grade 5), as it is common in the aerospace industry, its material properties can be found in Table 4.

***Table 4:*** Dimensions and material properties [17] of the trusses

| | |
|---|---|
| Outer diameter, $D_{OD}$ [in] | 0.5 |
| Thickness [in] | 0.049 |
| Inner diameter, $D_{ID}$ [in] | 0.402 |
| Cross-sectional area, A [$in^2$] | 0.07 |
| Elastic modulus, E [ksi] | 16 500 |
| Poisson ratio | 0.342 |
| Ultimate tensile strength, UTS [psi] | 138 000 |

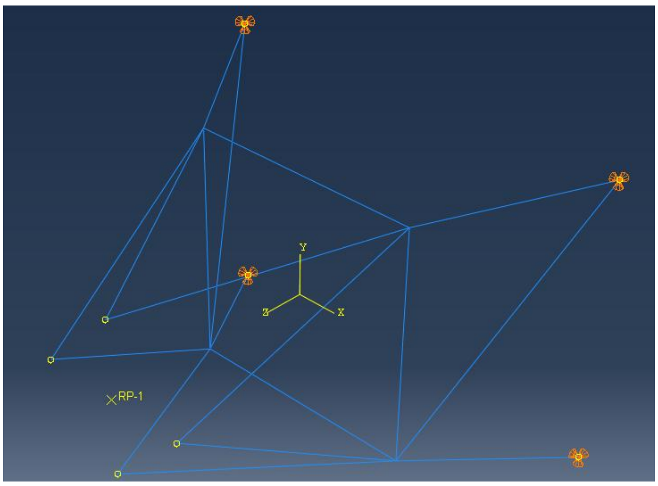


***Figure 5:*** Engine mount modelled in Abaqus

The bulkhead attachments were pinned to allow for rotational motion and model a more conservative setup. The weights of the motor and propeller were applied at the middle of the front frame, which is set as the reference point with coordinates (0,0,18). Assuming rigid connections between the trusses, the load applied at the reference point can be transferred to the engine attachment points, and thus the rest of the engine mount. This was done using an MPC constraint, or a multiple point constraint, that connects the reference point to the engine attachments with rigid beams.

*Load description*

All loads were computed using [15] and the example calculations in [18].

1. Load factors from the flight manual

Different load factors are specified in the AA-1A flight manual. A load factor is the ratio of the lift force and the weight. Depending on whether the flaps are up or down, this value can change and can have a positive or a negative value. A positive value signifies that the gravitational force is amplified by the specified value, and a negative value represents an upward force, going against the gravitational force.

When the flaps are down, the positive load factor for the utility category is +4.4, and the negative one is -1.76, while when the flaps are down, the load factor is +3.5 [9]. Since the highest positive load factor is +4.4, only this value will be used for the calculations for the positive inertial load. The mass of the electric motor is 53.5 kg, while the propeller chosen has a mass of 10.66 kg, which gives a total mass of 64.16 kg. This mass can be used to calculate the static load, which is the following:

$$W = m_{total}g = (53.5 + 10.66) \times 9.81 = 629.40 \text{ N}$$

With the positive and negative load factors, the inertial forces that will need to be applied to the engine mount are the following. Both these forces will have to be considered separately to model both cases appropriately.

$$F_{+} = n_{+} \times W = 4.4 \times -629.40 = -2769.4 \text{ N (a)}$$
$$F_{-} = n_{-} \times W = -1.76 \times -629.40 = 1107.8 \text{ N (b)}$$

2. Engine torque

Per FAR 23.361, the engine mount must simultaneously support the maximum engine torque ($T_{max}$), which is given by the manufacturer to be 370 Nm, and withstand 75% of the limit loads computed above at take-off. During cruise conditions, the full limit load must be considered with the maximum torque. This torque must be multiplied by a factor of 1.6 to account for propeller control system failure and by 1.25 to account for a turbopropeller installation. This gives a torque value of:

$$T = T_{max} \times 1.6 \times 1.25 = 370 \times 1.6 \times 1.25 = 740 \text{ Nm}$$

For both the take-off and the cruise cases, side loads do not need to be considered.

3. Side loads

According to FAR 23.363, a lateral load needs to be applied to the engine mount, equivalent to the static load multiplied by a load factor of 1.33. This gives a lateral force of:

$$F_x = n \times W = 1.33 \times -629.4 = -837.1 \text{ N}$$

4. Gyroscopic and aerodynamic loads

According to FAR 23.371, the engine mount needs to support the gyroscopic and aerodynamic loads induced by the rotating motor. The gyroscopic moment due to yaw or pitch can be calculated using the following equation:

$$M_{gyro} = I \times (\omega k \times \Omega)$$

Where $I$ is the mass moment of inertia of the propeller, $\omega$ is the angular velocity of the propeller (taken in the clockwise direction), and $\Omega$ is either the yaw or pitch rate in the z and y direction, respectively.

For these cases, a load factor of 2.5 on the static load must be considered, along with a yaw rate of 2.5 rad/s and a pitch rate of 1 rad/s, considered separately to find the two moments that are induced by the yawing and pitching motions. The inertial force is equivalent to 1573.5 N, using the load factor stated above. This is either an upwards or a downwards force, depending on which cases are considered.

The inertia of the propeller is $I = \frac{1}{12} m_{propeller} D^2 = \frac{1}{12}(10.66)(1.8034)^2 = 2.9 \text{ kg m}^2$.

The propeller RPM is taken to be the maximum RPM of the electric motor, which is 6500 RPM, or 680.7 rad/s. Using the coordinate system displayed in Fig. 5, the moment about the z-axis is the maximum torque from the motor, which is 740 Nm. As for the moment about the x and y axes, they can be computed as follows:

Using the pitch rate of 1 rad/s, a moment in y is introduced: $M_y = 2.9 \times 680.7 \times 1 = 1966.4$ Nm.

Using the yaw rate of 2.5 rad/s, a moment in x is introduced: $M_x = 2.9 \times 680.7 \times 2.5 = 4916.1$ Nm.

<u>*Load cases*</u>

Multiple load setups will be used to analyze the engine mount. All loads were multiplied by a factor of safety of 1.5, as per FAR 23.303 and were applied at the center of the front frame. They were converted to pounds before applying them in Abaqus to keep consistent units. The maximum stress will need to be compared to the ultimate tensile strength of Titanium Ti-6Al-4V (Grade 5) to evaluate if the structure will fail under any of the load cases. All results from the Abaqus analysis can be seen in the Annex.

**Load case 1a and 1b**: (a) The maximum augmented engine torque (from FAR 23.361), which is in the negative x direction for a clockwise propeller rotation, occurred when 100% of the positive inertial load was applied. This corresponds to the cruise conditions (i.e. maximum continuous conditions). In order to test the worst-case configuration, only the cruise configuration is used and not the take-off condition, since only 75% of the inertial loads are needed at take-off. (b) The same configuration is used but with the maximum negative inertial load.

**Load case 2**: The static load and the side loads are considered. The side loads are not included in the other load cases because, according to FAR regulations, they can be considered separately.

**Load case 3a and 3b**: (a) The maximum augmented engine torque (from FAR 23.361), in addition to yaw right, and pitch down moments are considered, along with the 2.5g inertial load downwards that is induced by these moments. (b) The same configuration is used but with yaw left and pitch up moments that introduce a 2.5g inertial load upwards.

A summary of all the load cases with the different loads and moments applied can be found in Table 5.

***Table 5:*** Forces and moments applied for each load case

| Case | Forces applied | Moments applied |
|---|---|---|
| 1a | $F_y = 1.5 \times (-2769.4) = -4154.1$ N | $M_z = 1.5 \times -740 = -1110$ Nm |
| 1b | $F_y = 1.5 \times (1107.8) = 944.1$ N | $M_z = 1.5 \times -740 = -1110$ Nm |
| 2 | $F_x = 1.5 \times -837.1 = -1255.7$ N<br>$F_y = 1.5 \times -629.4 = -944.1$ N | None |
| 3a | $F_y = 1.5 \times (-1573.5) = -2360.3$ N | $M_x = 1.5 \times -4916.1 = -7374.1$Nm<br>$M_y = 1.5 \times -1966.4 = -2949.6$ Nm<br>$M_z = 1.5 \times 740 = -1110$ Nm |
| 3b | $F_y = 1.5 \times (1573.5) = 2360.3$ N | $M_x = 1.5 \times 4916.1 = 7374.1$ Nm<br>$M_y = 1.5 \times 1966.4 = 2949.6$ Nm<br>$M_z = 1.5 \times 740 = -1110$ Nm |

*Results*

The maximum tensile and compressive normal stresses from each load case are summarized in Table 6. The maximum normal stresses are used to visualize the tension and compression in the truss members along their axes. This was done with the S11 function in Abaqus.

***Table 6:*** Maximum stresses in each load case

| Case | Maximum tensile stress (psi) | Maximum compressive stress (psi) | Safety factor |
|---|---|---|---|
| 1a | 14520 | -16430 | 8.4 |
| 1b | 10570 | -10030 | 13.1 |
| 2 | 5747 | -6156 | 22.4 |
| 3a | 80670 | -59580 | 1.7 |
| 3b | 67690 | -85910 | 1.6 |

For the safety factor, the maximum absolute value between the tensile and the compressive normal stresses was taken compared to the ultimate tensile strength of Titanium. The lowest safety factor is for Case 3b, which had the gyroscopic loads applied. An important thing to note is that since all loads were premultiplied by 1.5, all the safety factors are higher and are satisfactory per FAR 23.303. All the maximum stresses are below the ultimate tensile strength of titanium. Thus, this ensures that the engine mount will withstand the loads during flight and takeoff.

*Buckling analysis*

Now that the loads have been considered, a buckling analysis must be completed using the worst load case, which is case 3b. To do so, the highest compressive stress needs to be compared to Euler's critical buckling to ensure that the truss members do not buckle. The critical buckling stress is given by [19]:

$$\sigma_{cr} = \frac{\pi^2 EI}{A(KL)^2}$$

Where $I$ is the moment of inertia of a hollow shaft, given by $I = \frac{\pi(D_{OD}^4 - D_{ID}^4)}{64}$ [20], A is the cross-sectional area, K is a factor that depends on the boundary conditions of the members, and L is the length of the evaluated member, which is 20.89" here. Assuming a pinned-pinned member, K=1:

$$\sigma_{cr} = \frac{\pi^2(16\,500\ \times 10^3 \times 0.002)}{0.07(20.88)^2} = 9603 \text{ psi}$$

This value is much lower than the maximum compressive stress from Case 3b, however, this stress is taken at that specific node and is not representative. Taking the compressive stress in the middle of the member, which is 9113 psi, would be more representative. Comparing these two values, the structure should not buckle as its maximum compressive stress is below the critical buckling stress. However, since the values are close, the structure could be fortified the structure by adding supports.

## 4. Performance analysis

*Total Usable Energy*

We calculate the performance with a maximum take-off weight W = 1422 lbs (645 kg), 200kW motor (150 kW continuous), four batteries with $E_{mod} = 7101\,Wh$. We find a propulsion-chain efficiency of $\eta = 0.98 \text{ x } 0.95 \text{ x } 0.80 \sim 0.73$ (0.73 stays conservative), which is the combination of the inverter, motor, and

propeller efficiencies, respectively [21]. We also use the usable energy fraction baseline $f_{usable} = 0.8$ to preserve pack life, margin and safety.
Thus, the total and usable battery is as follows:

$$E_{tot} = 4E_{mod} = 28.404 \text{ kWh}, \qquad E_{usable} = f_{usable}E_{tot} = 22.72 \text{ kWh}$$

*Take-off*

We assume liftoff $V_{LOF} = 65 \text{ mph} = 29.1 \text{ m/s}$, ground roll $= 900\text{ft} = 274.3 \text{ m}$, and the average ground speed $\bar{V} = 0.6V_{LOF} \rightarrow t_{TO} = \frac{s}{\bar{V}} = 15.7s = 0.00437 \text{ h}$ [9, 22, 23, 24]. We also find shaft power = 80 kW, thus electrical input $P_{elec} = \frac{80}{\eta} = 109.6$ kW. From this, we find:

$$E_{takeoff} = P_{elec}t_{takeoff} = 0.48 \text{ kWh}$$

$$R_{takeoff} = \frac{900}{6076.12} = 0.148 \text{ NM}$$

*Climb*

Using the overall electrical-to-mechanical efficiency η, we use the following parameters to calculate the total climb energy:

- $m = 645$ kg
- $g = 9.81 \text{ m/s}^2$
- $h = 8000 \text{ ft} = 2438.4 \text{ m}$ (per POH manual)
- $3.6 \, x \, 10^6$ (factor to convert joules to kWh)

$$E_{climb} = \frac{mgh}{\eta(3.6 \times 10^6)} = \frac{645 \times 9.81 \times 2438.4}{0.73 \times 3.6 \times 10^6} = 5.87 \text{ kWh}$$

We also find the duration of time using values according to the POH:

***Table 7:*** Altitude and corresponding climbing rates [9]

| **Altitude (ft)** | **Climbing Rate (ft/min)** |
|---|---|
| 0 (Sea Level) | 765 |
| 2500 | 645 |
| 4500 | 550 |
| 6500 | 450 |

Thus, the time to get to altitude of 8000ft [9]:

$$t_{climb} = \frac{2500}{765} + \frac{4500 - 2500}{645} + \frac{6500 - 4500}{550} + \frac{8000 - 6500}{450}$$

$$t_{climb} = 13.34 \text{ min} = 0.22233 \text{ hr}$$

Considering the POH value $V_{climb} = 95$ mph to calculate range, we find [9]:

$$V_{vertical} = \frac{h}{t} = \frac{8000}{13.34} = 599.7 \text{ ft/min} \rightarrow 599.7 \times \frac{60}{5280} = 6.82 \text{ mph}$$

$$V_{horizontal} = \sqrt{(V_{climb})^2 - (V_{vertical})^2} = \sqrt{(95)^2 - 6.82)^2} = 94.78 \text{ mph} \rightarrow \frac{94.78}{1.15078} = 82.36 \text{ kt.}$$

$$R_{climb} = V_{horizontal} t_{climb} = 94.78 \times 0.22233 = 21.07 \text{ mi} \rightarrow \frac{21.07}{1.15078} = 18.31 \text{ NM}$$

*Descent*

The descent considers a slow descent at a rate $w = 600$ ft/min (consistent with common 3° descent) [25].

$$t_{descent} = \frac{8000}{600} = 13.3 \text{ min} = 0.222 \text{ hr}$$

The avionics load $P_{avionics} = 1.5$ kW, thus:

$$E_{descent} = P_{avionics} t_{descent} = 0.33 \text{ kWh}$$

$$R_{descent} = V_{cruise} t_{climb} = 117\ x\ 0.222 = 26.0 \text{ NM}$$

*Reserve*

We get $P_{cruise}$ from the gas engine Lycoming O-235 that produces 108 hp at max power. Cruise is typically 65-75% of max power for small aircraft, thus conservatively $P_{cruise} = 0.75 \times 108 = 81\ hp \rightarrow 60\ kW$. We chose a 5-minute reserve since our design cannot satisfy the conventional 30-minute VFR fuel-reserve requirements:

$$E_{reserve,30} = 60 \times 0.5\text{hr} = 30 \text{ kWh} \rightarrow 30 \text{ kWh} > E_{usable}\ (= 22.7 \text{ kWh})$$

The 5-minute (0.083hr) reserve would only be a project-level demonstration reserve used for short local test flights.

$$E_{reserve} = P_{reserve} t_{reserve} = 60 \times 0.083 = 5 \text{ kWh} \rightarrow 5 \text{ kWh} < E_{usable}$$

$$R_{reserve} = V_{cruise} t_{reserve} = 117 \times 0.083 = 9.75 \text{ NM}$$

*Cruise*

Using the energies taken during each necessary phase, we find that the remaining energy is used for cruising:

$$E_{cruise} = E_{usable} - E_{takeoff} - E_{climb} - E_{descent} - E_{reserve}$$

$$E_{cruise} = 22.72 - 0.48 - 5.87 - 0.33 - 5.00 = 11.04 \text{ kWh}$$

The POH lists cruise speeds at 8000ft of 134 mph at 75% power and 123 mph at 65% power for the standard AA-1 Yankee, thus we adopt:

$$t_{cruise} = \frac{E_{cruise}}{P_{cruise}} = \frac{11.04}{60} = 0.184 \text{ h (11.0 mins)}$$

$$V_{cruise} = 134 \text{ mph} = 117 \text{ kt}$$

$$R_{cruise} = t_{cruise} \times V_{cruise} = 0.184 \times 117 = 21.5 \text{ NM}$$

Thus, under these conservative assumptions, the electrified AA-1 can sustain roughly 11 minutes of cruise beyond the top of the climb and cover roughly 21.5 NM before the planned reserve and descent are reached.

*Performance Conclusion*

We find that the totals for time and range in air are:

$$t_{total} = t_{takeoff} + t_{climb} + t_{cruise} + t_{descent} + t_{reserve}$$

$$t_{total} = 0.26 + 13.34 + 11.0 + 13.3 + 5.0 = 42.9 \text{ minutes}$$

$$R_{total} = R_{takeoff} + R_{climb} + R_{cruise} + R_{descent} + R_{reserve}$$

$$R_{total} = 0.148 + 18.3 + 21.5 + 26.0 + 9.75 = 75.75\ NM \sim 87.17 \text{ miles}$$

We also consider the energy intensity per nautical mile [26], where:

$$I_{cruise} = \frac{E_{cruise}}{R_{cruise}} = \frac{11.52}{21.50} = 0.513 \text{ kWh/NM}$$

$$I_{mission} = \frac{E_{usable}}{R_{total}} = \frac{22.72}{75.75} = 0.300 \text{ kWh/NM}$$

This means that each additional cruise nautical mile costs 0.513 kWh. Here, the mission intensity is lower than cruise-only due to the additional descent distance with minimal energy. Using intensity to compare with the fuel AA-1A: the electrified configuration delivers 75.75 NM total range, while the fuel AA-1A achieves 381.5 NM. At the cruise-only intensity, matching the AA-1A's distance would require 381.5 x 0.513 = 196 kWh just for cruise energy, which is much beyond the current total usable energy of 22.72 kWh. One way to improve this would be to lower the required cruise power and thus reduce $I_{cruise}$. This can be done by flying slower near the best-range speed, using a prop matched for the cruise, and basic drag clean-up such as wheel fairings, gap seals, and smooth tape on seams.

## 5. Avionics

*Electrical Wiring Modifications*

With this new system, the power, battery charge level and temperature of the motor will have to be monitored. Thus, the aircraft's wiring diagram must be modified to include these new sensors and the electric motor, batteries and controllers. The avionics must include a monitoring system to oversee all these components.

Since our aircraft has been modified to include an electric motor instead of a gas-powered engine, some components of the electrical system diagram must be removed. To accommodate our electric motor within the existing electrical systems diagram of an AA1A Yankee Clipper, we will be dividing our electrical system into 2 subsystems: high voltage and low voltage. The high-voltage system will be responsible for powering the electrical motor and propeller from the four Mobius Husky batteries through the use of a contactor and a motor controller. Meanwhile, the low-voltage system, powered by a 12V battery, will power the general electrical bus, the dome light, clock, and hour meter. A very high voltage being supplied by the Mobius batteries requires our systems to have a DC-to-DC converter.

Therefore, a list of components to be removed from the electrical system has been determined, with all the components necessary for the gas-powered engine being taken out as they don't serve any purpose for an electrical motor. The starter switch, starter motor, and starter solenoid were used to crank the internal combustion engine during start-up and are not useful anymore. The alternator was used to generate electricity from the engine's produced energy [27]. However, the new electrical diagram has a dedicated battery pack, making the alternator obsolete. The transistorized regulator served to regulate the voltage coming out of the alternator; as there is no longer an alternator, the regulator can be pulled out [28]. Finally, the sensors related to oil and fuel can be removed because the redesigned plane doesn't require them.

The components added, on the other hand, all serve a purpose. It can be for safety reasons, power reasons, or else, but all are useful for the plane's ability to fly. The obvious additions are the batteries and the electric motor, without which the plane wouldn't be able to fly. One powers the aircraft, and the other provides the propulsion. The controller serves as the relay between the battery pack and the motor. It regulates power delivery, the motor speed, and provides other protections useful to the system. The AAMCi – Series V2 800 module is added to safely connect and disconnect the electric loads [29]. With its capacity to manage up to 800V with DC, it is ideal for the Mobius Husky batteries. The DC-to-DC converter is a key component for the low-voltage bus, as it is the port of entry for the subsystem. Its role is to step down the voltage for avionics and take the alternator's role [30]. The hourmeter and clock are required to maintain standard cockpit instrumentation and flight time as per the AA1B POH [31]. Two master switches are present for independent control of the low and high-voltage subsystems. Finally, the battery management system (BMS) and the controller area network (CAN) were added. Their purposes are explained in the battery management system subsection.
The modifications made to the system can be seen in the following wiring diagrams (Figures 6 and 7). The modified wiring diagram (Figure 7) was created with the help of the AA1A and AA1B POHs.

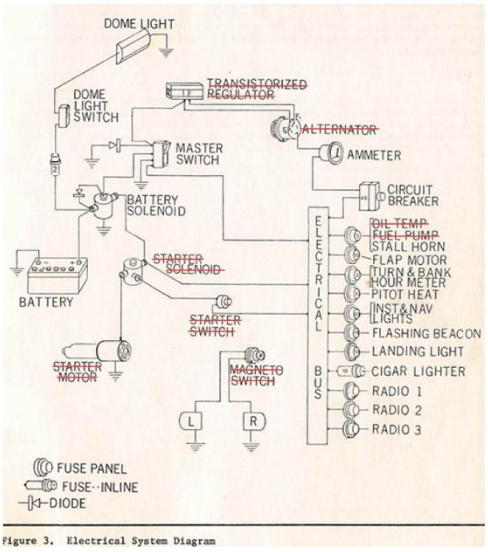


***Fig 6:*** Wiring diagram from POH [9] with removed components

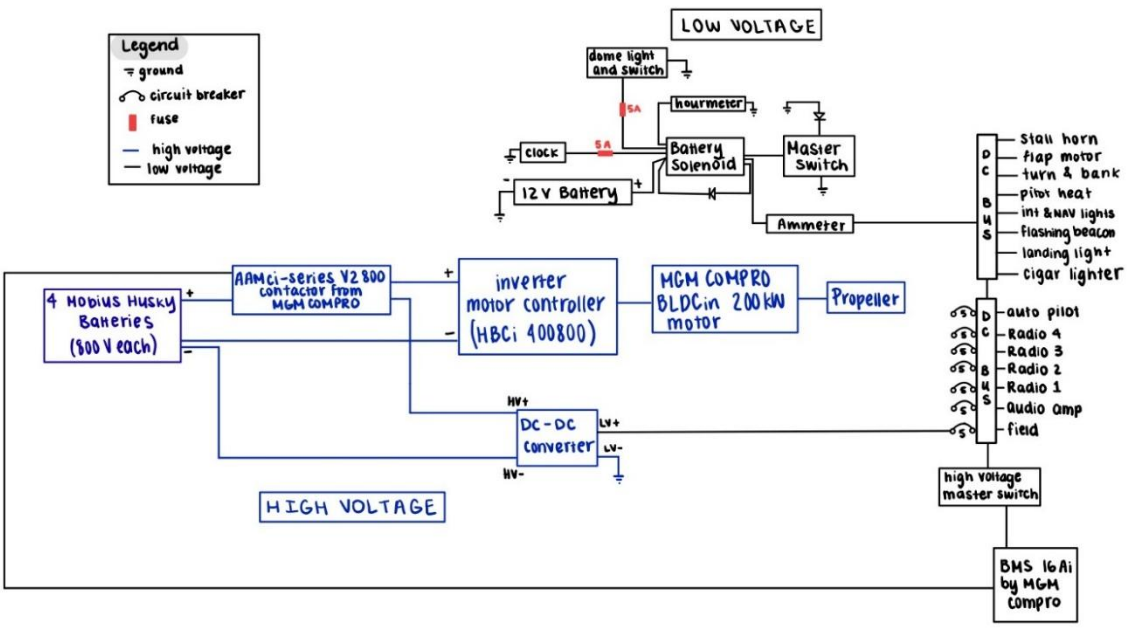


***Fig 7:*** Modified Wiring diagram

*Motor monitoring*

A motor monitoring system is required to monitor the motor's RPM, power output, and status, and relay all the parameters to the pilot during flight on the display panel, ensuring proper operation, which will extend the lifespan of the motor. The motor controllers accomplish part of the monitoring already, but additional features are required.
Other parameters that need to be monitored include the motor vibrations, which can be used to evaluate the strength and performance of the engine mount, the temperature to ensure that it is kept within the operating range, and the current and voltage of the motor.
These values can be used to optimize the performance of the motor and for maintenance, but also to ensure that the system is well preserved.

*Battery management system*

A BMS is required to monitor the health of the batteries and maintain them operational over a long period. The BMS 16Ai [32], offered by MGM COMPRO, is a good option since it operates at a maximum voltage of 800 V, which is the maximum that can be provided by the batteries.
The BMS offers multiple beneficial features:

- It measures the batteries' temperature to ensure it remains within operational range. Thus, the thermal monitoring system is included in the BMS.
- It monitors the voltage and current outputs during charging and operation, ensuring proper function.
- It ensures a balance between the different battery cells to maximize the capacity of the battery
- It has low power consumption, is lightweight, and is fully independent.

This data will be transmitted to the pilot to ensure that all these values are known and monitored. To ensure communication is fluid, as reliable as possible, and stored, a CAN system is meant to be deployed. It is a data bus that allows data transmission without needing a central computer [33].

*Panel Redesign*
The instrument panel had to be redesigned to adapt to the new electrical system. Most components removed (crossed in red) were replaced by different functionalities. Only the ignition switch was completely removed, and its assigned number was given to a new component. Here are the new roles for components 9, 13, 21, 22, 23, 24, 27, 28, and 29:

- 9: Electric RPM Indicator
- 13: Digital Clock
- 21: System Current Display
- 22: System Voltage Display
- 23: Power Output Display
- 24: Batteries Internal Heat Display
- 27: High-Voltage Master Switch
- 28: Low-Voltage Master Switch
- 29: Batteries States of Charge

The components 16, 17, and 26 do not need to be replaced but rewired to fit the new electrical system, hence the yellow sign next to them. A sketch of the new panel versus the old panel can be seen in Figure 8.

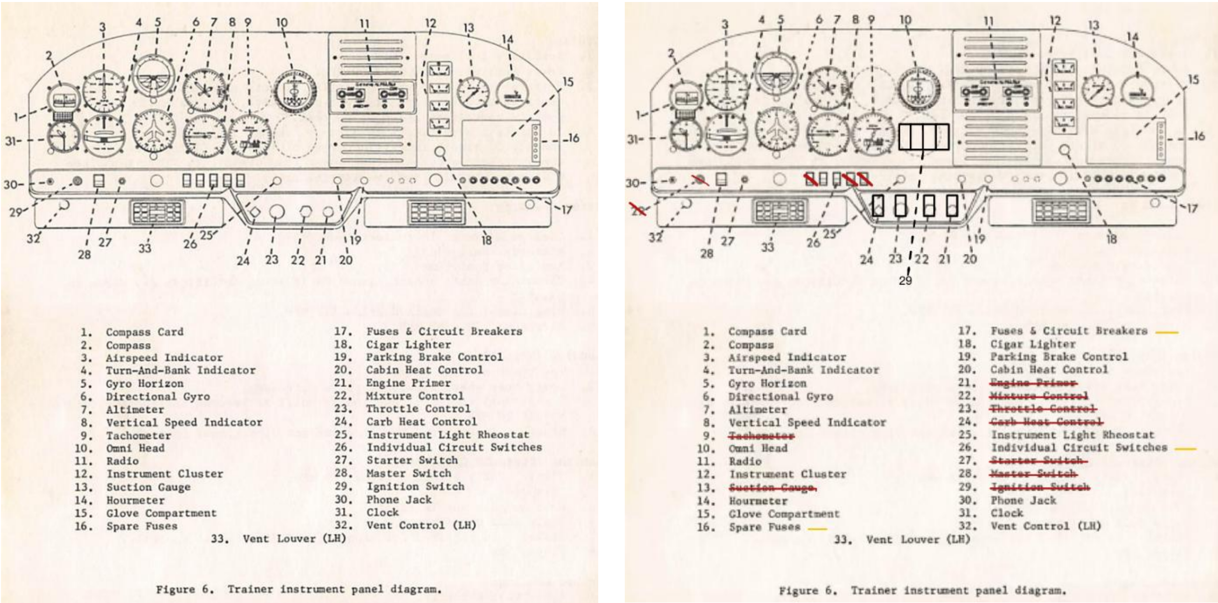


***Fig 8:*** Old [9] vs New panel

## 6. Conclusion

The electrification of the AA-1A Yankee Clipper was evaluated in this report. Modifications were made to accommodate the new electric propulsion system, including the design and analysis of a new engine mount and a modified avionics system that included all necessary components needed to monitor and operate the system. While the implementation of an electric motor, batteries and electronics system is possible, it was found that compared to a gas turbine engine, the performance of the aircraft is far reduced. This is due to the added weight from the batteries, which is not lost by the aircraft during flight like fuel is on a regular aircraft. It is also linked to the low power density that batteries provide compared to fuel. To achieve the same range, more batteries are required, which brings more weight and reduces the overall performance since the weight of the aircraft modifies the performance and range that can be achieved. While this is an interesting alternative to gas turbine engines, implementing a fully electric propulsion system is not feasible with the resources available currently, from a monetary and an efficiency point of view. Nowadays, most companies are exploring a hybrid propulsion system instead, to extract the benefits from each system while still progressing towards a greener solution, better for the environment and for people. A fully electric propulsion system could be possible if a better power density could be achieved.

## 7. Annex + References

# Annex

*Results from the load cases*

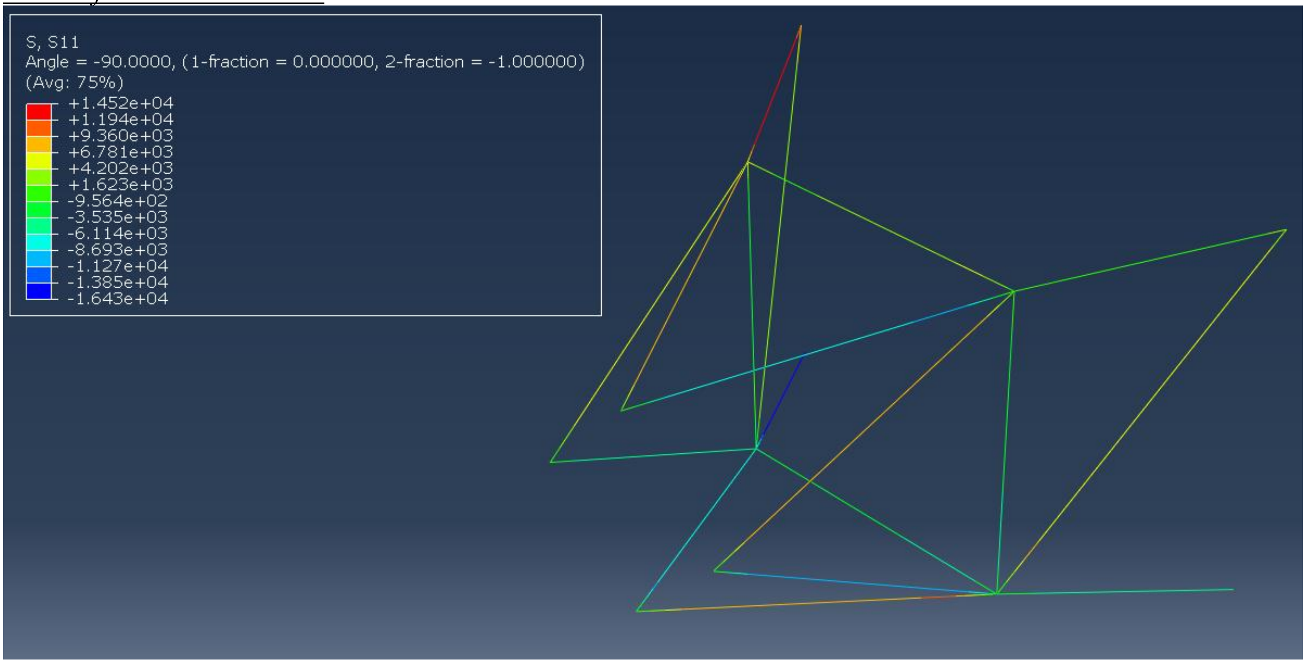


***Figure 9:*** Load case 1a

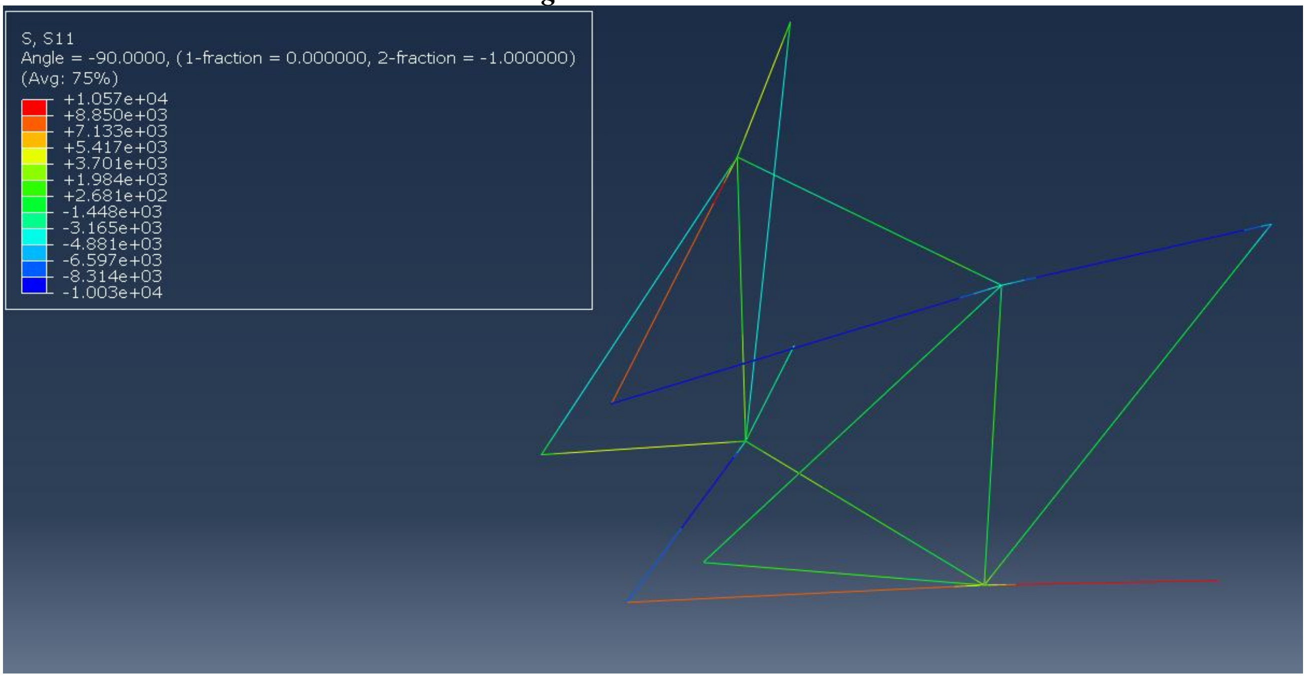


***Figure 10:*** Load case 1b

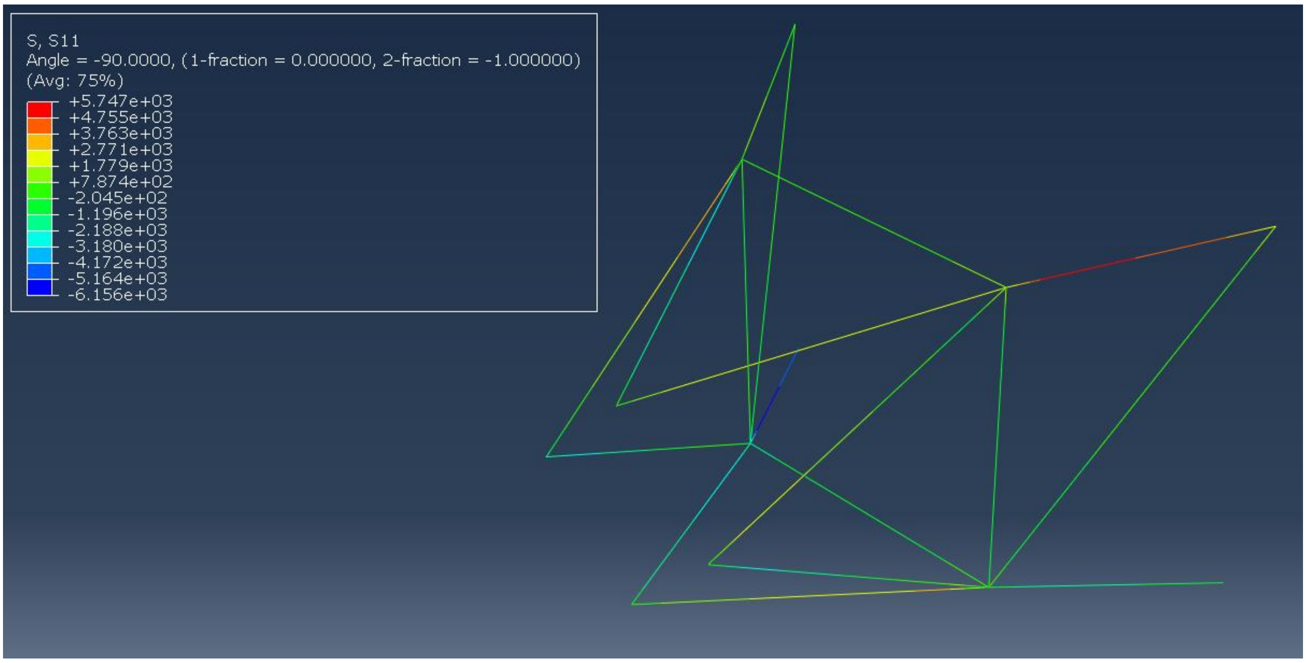


***Figure 11:*** Load case 2

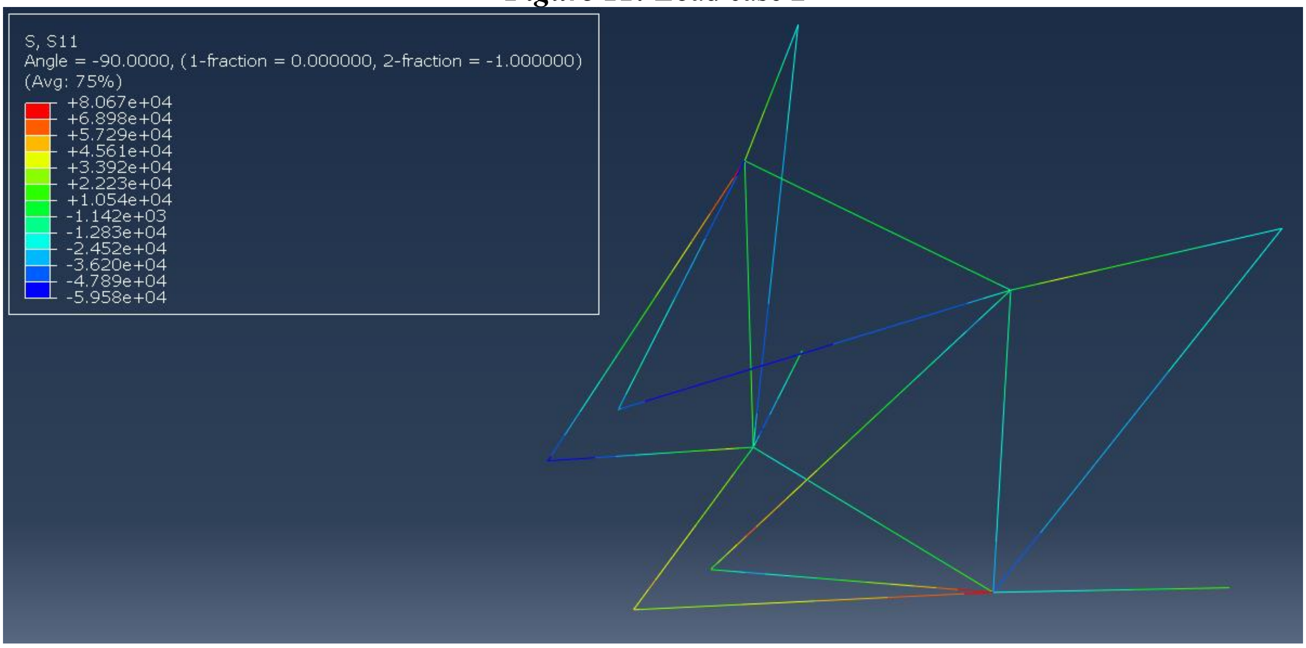


***Figure 12:*** Load case 3a

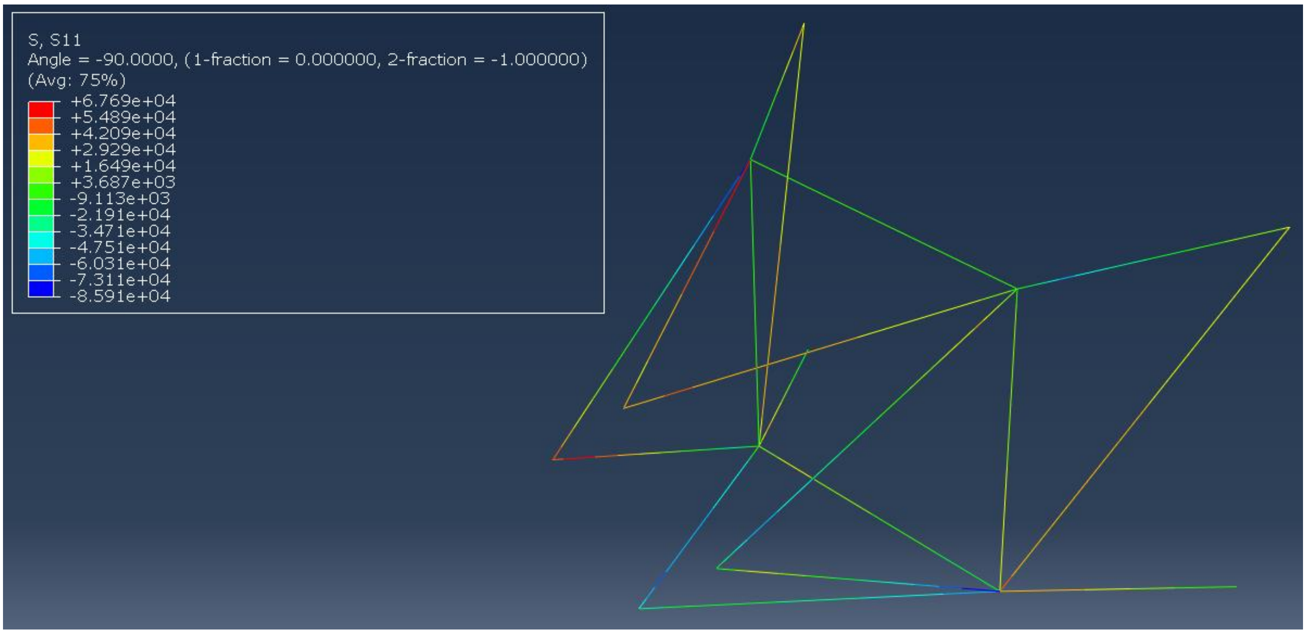


***Figure 13:*** Load case 3b

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