Design of a Short to Medium Range

Hybrid Transport Aircraft

A Project Presented to
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by

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approved by

[Signature]

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Faculty Advisor
Design of a Hybrid Transport Aircraft

**Nomenclature**

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<td>A</td>
<td>aspect ratio</td>
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<tr>
<td>A/C</td>
<td>aircraft</td>
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<tr>
<td>AEP</td>
<td>airplane estimated price</td>
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<tr>
<td>$A_h$</td>
<td>horizontal tail aspect ratio</td>
</tr>
<tr>
<td>$A_{int}$</td>
<td>capture area per inlet in square feet</td>
</tr>
<tr>
<td>$A_v$</td>
<td>vertical tail aspect ratio</td>
</tr>
<tr>
<td>ac</td>
<td>aerodynamic center</td>
</tr>
<tr>
<td>$AH_j$</td>
<td>number of flight hours per year for crew member j</td>
</tr>
<tr>
<td>ASP</td>
<td>avionics systems price</td>
</tr>
<tr>
<td>b</td>
<td>wingspan</td>
</tr>
<tr>
<td>B.L.</td>
<td>wing buttock lines</td>
</tr>
<tr>
<td>$C_L$</td>
<td>lift coefficient</td>
</tr>
<tr>
<td>$C_D$</td>
<td>drag coefficient</td>
</tr>
<tr>
<td>Cr</td>
<td>cruise</td>
</tr>
<tr>
<td>c</td>
<td>chord</td>
</tr>
<tr>
<td>$c_r$</td>
<td>root chord</td>
</tr>
<tr>
<td>$c_t$</td>
<td>tip chord</td>
</tr>
<tr>
<td>CGR</td>
<td>climb gradient</td>
</tr>
<tr>
<td>cg</td>
<td>center of gravity</td>
</tr>
<tr>
<td>$C_{n8r}$</td>
<td>control power derivative</td>
</tr>
<tr>
<td>$C_{lab/ap}$</td>
<td>labor cost of airframe and system maintenance</td>
</tr>
<tr>
<td>$C_{lab/eng}$</td>
<td>labor cost of engine maintenance</td>
</tr>
<tr>
<td>$C_{mat/ap}$</td>
<td>cost of maintenance materials for the airframe and systems</td>
</tr>
<tr>
<td>$C_{mat/eng}$</td>
<td>cost of maintenance materials for the engines</td>
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<tr>
<td>$C_{amb}$</td>
<td>applied maintenance burden</td>
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<tr>
<td>$C_{dap}$</td>
<td>cost of airplane depreciation</td>
</tr>
<tr>
<td>$C_{deng}$</td>
<td>cost of engine depreciation</td>
</tr>
<tr>
<td>$C_{dav}$</td>
<td>cost of avionics system depreciation</td>
</tr>
<tr>
<td>$C_{dapsp}$</td>
<td>cost of airplane spare parts depreciation</td>
</tr>
<tr>
<td>$C_{dengsp}$</td>
<td>cost of engine spare parts depreciation</td>
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<tr>
<td>$C_{lf}$</td>
<td>direct operating cost due to landing fees</td>
</tr>
<tr>
<td>$C_{apl/f}$</td>
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<tr>
<td>$C_{nf}$</td>
<td>navigation fee</td>
</tr>
<tr>
<td>$C_{apnf}$</td>
<td>navigation fee charged per airplane per flight</td>
</tr>
<tr>
<td>$C_{rt}$</td>
<td>direct cost of registry taxes</td>
</tr>
<tr>
<td>DOC</td>
<td>direct operating cost</td>
</tr>
<tr>
<td>$DOC_{lnr}$</td>
<td>direct operating cost of landing fees</td>
</tr>
<tr>
<td>$DOC_{fin}$</td>
<td>direct operating cost of financing the airplane</td>
</tr>
<tr>
<td>DP$_{ap}$</td>
<td>airplane depreciation period</td>
</tr>
<tr>
<td>DP$_{av}$</td>
<td>airplane avionics system depreciation period</td>
</tr>
<tr>
<td>DP$_{engsp}$</td>
<td>depreciation period for engine spare parts</td>
</tr>
<tr>
<td>e</td>
<td>battery specific energy, Oswald constant</td>
</tr>
<tr>
<td>$E_{bat}$</td>
<td>energy of batteries</td>
</tr>
<tr>
<td>$E_{\tau}$</td>
<td>propulsive energy required</td>
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EP = engine price
ESPPF = engine spare parts price factor
$F_{\text{dap}}$ = airframe depreciation factor
$F_{\text{dav}}$ = airplane avionics system depreciation factor
$f$ = equivalent parasite area
F.S. = fuselage stations
FAA = Federal Aviation Administration
FAR = Federal Aviation Regulations
FT = Fischer-Tropsch
$F_{\text{engsp}}$ = engine spare parts factor
$f_{\text{rt}}$ = factor depending on airplane size
g = gravitational acceleration
$h$ = height
$h_{\text{fus}}$ = height of fuselage in feet
HTA = Hybrid Transport Aircraft
$i_{\text{w}}$ = incidence angle
$j$ = crew member type
$K$ = drag due to lift polar coefficient
$K_g$ = gust alleviation factor
$k_j$ = factor accounting for crew vacation pay, cost of training crew insurance, etc.
LE = leading edge
LG = landing gear
Ltr = loiter
$l_{\text{fus}}$ = length of fuselage in feet
$l_h$ = distance from wing root quarter cord to horizontal tail quarter cord in ft
$l_n$ = nacelle length
$n$ = load factor
$n_{c_j}$ = number of crew members of type $j$
$N_D$ = drag induced yawing moment
$n_e$ = number of engines
$n_{\text{limpos}}$ = positive limit maneuvering load factor
$n_{\text{limneg}}$ = negative limit maneuvering load factor
$N_{\text{int}}$ = number of inlets
$N_{\text{pax}}$ = number of passengers
$N_{\text{pil}}$ = number of pilots
mgc = mean geometric chord
$M_H$ = maximum Mach number at sea level
$M_{\text{ff}}$ = mission fuel fraction
$M_{\text{res}}$ = reserve fuel fraction
$M_{\text{tfo}}$ = trapped fuel fraction
MEA = More Electric Aircraft
$p$ = battery specific power
$P_r$ = power required (propulsive)
$P_2$ = maximum static pressure at engine compressor face in psi
$q$ = dynamic pressure in psf
$q_d$ = design dive dynamic pressure in psf
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RC = rate of climb
Re_r = Reynold’s number at root chord
Re_t = Reynold’s number at chord tip
R_{bl} = block distance
R_{cl} = climb distance
R_{de} = descent distance
R_{man} = maneuvering distance
S = wing reference surface area
S_h = horizontal tail area
S_v = vertical tail area
S = gross wing area
SAL_j = yearly salary per crew member
S_{TOFL} = takeoff field length
T = time of flight
T_{phase} = time of flight for a certain phase
TBW = Truss Braced Wing
t/c = thickness ratio
T_{cl} = time to climb
T/W = thrust loading (or thrust to weight ratio)
t_{rh} = horizontal tail maximum root thickness
t_{rv} = vertical tail maximum root thickness
TEF_j = travel expense factor per crew member j
W/S = Wing loading
W_{ai} = air induction system weight
W_{api} = weight of air-conditioning, pressurization, anti-icing and de-icing system
W_{apu} = weight of auxiliary power unit (APU)
W_{bat} = weight of batteries
W_{bc} = weight of baggage and cargo handling equipment
W_e = empty weight
W_{els} = weight of electrical system
W_{eng} = weight of engine
W_f = fuel weight
W_{fc} = weight of flight control system
W_{fur} = weight of furnishings
W_{hs} = weight of hydraulic and pneumatic system
W_i = weight of instruments
W_{iae} = weight of the instrumentation, avionics, and electronics
W_{oe} = operating empty weight
W_{ops} = weight of operational items
W_{ox} = weight of oxygen system
W_ppl = payload weight
W_{pt} = weight of paint
W_{f0} = trapped fuel weight
W_{to} = take off weight
W_L = landing weight
W_{fs} = fuel system weight
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\[ W_{fus} = \text{weight of fuselage in lbs} \]
\[ w_{fus} = \text{width of fuselage} \]
\[ W_p = \text{propulsion system} \]
\[ W_{pwr} = \text{powerplant weight} \]
\[ W_{struct} = \text{weight of structure} \]
\[ U_{ann bi} = \text{annual utilization in block hours} \]
\[ V = \text{velocity} \]
\[ V_B = \text{design speed for maximum gust intensity} \]
\[ V_A = \text{approach speed} \]
\[ V_C = \text{design cruise speed} \]
\[ V_{cr} = \text{cruise velocity} \]
\[ V_D = \text{design diving speed} \]
\[ V_M = \text{design maneuvering speed} \]
\[ V_{pax} = \text{passenger cabin volume} \]
\[ V_S = +1g \text{ stall speed or minimum speed at which airplane is controllable} \]
\[ x_p, x_h, x_c = \text{distance from center of gravity to aerodynamic center of the surface} \]
\[ y_t = \text{lateral thrust moment arm of most critical engine} \]
\[ z_h = \text{distance from vertical tail to root, where horizontal tail is mounted on vertical tail} \]
\[ \eta_p = \text{propeller efficiency} \]
\[ \rho = \text{air density} \]
\[ \Lambda = \text{quarter chord sweep angle} \]
\[ \lambda = \text{taper ratio} \]
\[ \varepsilon_w = \text{twist angle} \]
\[ \Gamma_w = \text{dihedral angle} \]
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1. The
The
The
While
As
1.1.
1.1.1.
With

Conventional propulsion systems thermally decompose oil in piston engines or gas turbines. This in turn produces mechanical power which runs either a turbofan or turbopropeller. Gearboxes can increase efficiency when they decouple the fan from the turbine or turbofan engine. Despite the ability to increase efficiency, these propulsion systems emit exhaust.

Figure 1. World Carbon Dioxide Emissions

Both 70%
Error! Reference source not found., illustrates the billions of metric tons of CO$_2$ that are emitted annually by each country worldwide. CO$_2$ has a lifetime of up to 200 years and has a global warming potential direct effect of 100 years, so its annual contribution effects are present for decades to come. Transportation produces approximately 31.8% of these emissions, contributing to greenhouse gases. (Davis, Williams, & Boundy, 2016)

Error! Reference source not found. summarizes the types of fuel emissions in pounds per gallon that contribute to direct CO$_2$ emissions. Aviation gasoline, or Av-gas, contributes 11% and jet fuel contributes 12%. (Davis, Williams, & Boundy, 2016) With an increase in the number of general aviation aircraft in use, these emissions will not decrease without changes to aircraft or fuel. General aviation consist of aircraft following general operating and flight rules, aircraft with a minimum seating capacity of 20 or maximum payload of 6,000 lbs that are not for hire, agricultural aircraft, rotorcraft external load operators, and commuter or on-demand aircraft not covered under FAR regulations part 121. From 1970-2014, the number of general aviation aircraft increased by around 1% per year, as summarized in Error! Reference source not found.. As a result in the number of increased aircraft, the energy used has increased by about 2% per year and 2.3% from 2004-2014. (Davis, Williams, & Boundy, 2016)
Besides producing a high volume of pollutants, aircraft also contribute to the use of fossil fuels. The United States imported 564,000,000 tons of oil in 2008 and has since increased oil importation. Transportation used a significant portion of

**Figure 3.** Number of General Aviation Aircraft from 1970 to 2014

Although

1.2.2.

Another consideration is the candidate for hydrocarbon feedstock. If some form of plants were used, there would need to be adequate farmland to sustain the fuel. This farmland would need to be acquired without deforestation techniques, where the smoke from fires would contribute to greenhouse gases and increase pollution. Algae holds promise as a future option due to its oil production. A comparison of algae to other feedstocks is present in Figure 4 and depicts the significantly higher oil production as compared to other biofuels. It is estimated to produce 10,000-20,000 gallons per acre per year of oil, a dramatic increase from the use of soybean oil.

**Figure 4.** Alternative Biofuels
Logistically,

1.2.3. Alternative Fuel Comparisons

Since

1.2.4. Lithium Ion Batteries

Other

Lithium ion batteries have been utilized in many applications, including electric vehicles and electronics. These batteries are considered the most commercially available, power-dense batteries. Graphite carbon composes the anode of the batteries; while lithiated metal oxides like LiMO₂ and LiCoO₂ are used as cathodes. With charging, the cathode transforms into lithium ions, which then move towards the anode through lithium salts combining with external electrons. (Farhadi & Mohammed, 2016) Battery measurements consist of specific energy densities, measured in Wh/kg, volumetric energy densities, measured in Wh/liter, and specific power, measured in W/kg. Although Li-ion batteries are more costly, priced at $0.82 per Wh, compared to Ni-Cd and Pb-acid, at $0.42 and $0.28, respectively, they have the highest energy density offered, which means they are the smallest in both volume and weight for the amount of energy stored. Figure 5 illustrates the capabilities of Li-ion batteries’ specific energy, volumetric energy density, and specific power, and compares them against

*Figure 5. Comparison of different types of* Ni-Cd and Pb-acid batteries. (Tariq, Maswood, Gajanayake, & Gupta, 2017). Studies have stated that Li-ion battery energy densities are increasing 8-10% per year, with power doubling every 10 years. The maximum specific energy density value potentially being 5200 Wh/kg
Design of a Hybrid Transport Aircraft

(Ullman). Figure 6 illustrates a summary of efficiency, discharge time, operating temperature, and lifespan of Li-ion batteries compared to other energy storage technologies.

Lithium ion batteries are 95-99% efficient. They have lifespans of 5-20 years with long discharge times and a wide range of temperatures in which they can operate. Seal level temperatures are usually around 15°C and flight temperatures range from -44 °C to -56 °C at cruise altitude. Considerations could be made to protect the batteries in the cool atmosphere. The long discharge time makes these batteries ideal for transportation applications. High energy storage batteries with long discharge time lead to maximum system efficiency combined with minimal cost, weight, and volume. (Farhadi & Mohammed, 2016) With cost, weight, and volume being critical factors in aircraft design, lithium ion batteries demonstrate good promise statistically and have illustrated their capabilities through their use in small aircraft.

![Figure 6. Performance Comparison of Li-ion Batteries](image)

With the increased level of technology for batteries, designing and building light and general aviation aircraft is becoming a reality. Although initially costly in production, these aircraft will not rely on fuel and will therefore have immense savings in the future. Prices will also dwindle with increasing availability of technology and scale economy effects. Table 1 lists electric aircraft that have flown as of 2016. (Riboldi & Gualdoni, 2016) These aircraft were referenced to perform an electric aircraft sizing for a light or general aviation aircraft.

![Table 1. Preliminary database of Electric Aircraft](image)
1.3.

To perform the sizing of the battery of a small airplane designed for cross country flying or training, the weight was first considered. For an electric aircraft, the weight is a combination of the weight of the payload, batteries, electric motor, and the empty weight of the vehicle itself,

\[ W_{to} = W_e + W_{pl} + W_{bat} + W_m \]  

(1)

The mission profile consisted of the usual phases of take-off, climb, cruise, loiter, and land. Typical flight sizing, relies on fuel fractions to define the fuel necessary for each portion of the flight. (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017) Since the fuel would not change, and the weight would stay constant, this method of sizing would not be appropriate. Instead, a relationship between the weights and the mission requirements was examined. In electric aircraft, the most significant phases that affect the power are climbing, cruising, and loitering. The power required for climb can be calculated through Eq. \( P_{climb} = \)

\[ P_{climb} = W_{to}RC + \frac{1}{2} \rho v_{climb}^3 S_D c_{D} \text{ (2) relating air density during the climb, constant airspeed, wing reference surface, the value of the drag coefficient, and the rate of climb.} \]

\[ P_{climb} = W_{to}RC + \frac{1}{2} \rho v_{climb}^3 S_D c_{D} \text{ (2)} \]

Using

Error! Reference source not found.)

where
Design of a Hybrid Transport Aircraft

and the Oswald coefficient, $e$. With clean values of $K$ and $C_D$, the lift coefficient can be calculated using $CL = 2W_{top}V2S$ (3).

$$C_L = \frac{2W_{to}}{\rho V^2S} \quad (3)$$

The next factor for sizing is the energy required. This involves energy required to climb, cruise, loiter, and land. To estimate the energy required to climb, the time to climb and the power required for the climb are taken into consideration, where the time to climb is defined in

$$TTC = h_{cruise} \cdot RC \quad (4)$$

$$TTC = \frac{h_{cruise}}{RC} \quad (4)$$

and the energy required for climb is defined in the following equation. (Riboldi & Gualdoni, 2016)

$$E_{\text{climb}} = P_r^{\text{climb}}TTC \quad (5)$$

The energy required for cruise and loiter can be assumed to be the same. Different values will be calculated because the densities, velocities, and drag coefficients differ for the phases. (Riboldi & Gualdoni, 2016)

$$P_r = \frac{1}{2} \rho V^3 S C_D \quad (6)$$

From there, the energy required for each phase can be calculated using

$$E_{\text{phase}} = P_r^{\text{phase}}T_{\text{phase}} \quad (7)$$

where the time of each phase relies on the range of the phase and the cruise speed at that phase. (Riboldi & Gualdoni, 2016) Finally, the battery weight can be determined. With this calculation, the propulsive efficiency is assumed to be less than 100%, providing the batteries
and motor with a higher requirement. The mission profile battery weight can be calculated using the following equation. (Riboldi & Gualdoni, 2016)

\[
W_{bat,MP} = \frac{g}{\eta_p} \max \left\{ \frac{e^{\text{climb}} + e^{\text{cruise}} + e^{\text{loiter}}}{p_{\text{max}}}, \max \left\{ \frac{p_{\text{climb}} + p_{\text{cruise}} + p_{\text{loiter}}}{p} \right\} \right\}
\] (8)

This sizing examination is limited in that it applies to current technology and small aircraft. Sizing aircraft in other weight categories could prove to be more challenging, especially with relying on current battery technology. A solution to transferring this process to a larger aircraft, and utilizing current technology, could be to size a hybrid aircraft that could rely both on the reusable power of batteries and on fuel to compensate for the additional size and weight strain.

Hybrid aircraft are defined as aircraft where the propulsion is powered by more than one type of energy source (Thauvin, et al., 2016). Different fuel sources provide the ability to improve the aircraft’s performance, reduce fossil fuel reliance, reduce noise levels, and emit fewer pollutants. Forms of Electrical Energy Storage (EES) would benefit hybrid vehicles in their ability to store and use electric power, as needed.

Research was performed on an aircraft with twin-turbo propeller engines with 3500 thermal horsepower per engine, with the capabilities of traveling 400 nautical miles, at a ceiling of 20,000 ft. The aircraft was also designed using technology for 2035, which would provide batteries with higher energy densities (Thauvin, et al., 2016). This aircraft’s mission overview can be viewed in Table 2. Its mission profile with the comparison of down-sized gas powered turbine and standard conditions can be viewed in

Figure 7. Mission Profile Comparison of Sample Aircraft. The mission profile illustrates the capabilities of added power provided by an energy storage unit combined with a reduced engine. Also visible in the mission profile is added power sent to the storage unit during descent.
Energy recovering through the use of EES can be achieved through a few different mechanisms. The research on this specific aircraft and the energy recovery examined the energy gained through braking and through gravitational potential energy. The amount of energy recovered and stored through braking is fairly minimal. For the research aircraft, the equivalent amount of fuel saved is 2 kg, which is less than the amount of fuel needed to taxi. These results stemmed from assuming a gas turbine efficiency of 40%, and the braking system consisting of disc brakes, thrust reversers, and airbrakes. (Thauvin, et al., 2016) Producing the equivalent of 2 kg of fuel or 0.19% of total energy used during this mission seems insubstantial.

Table 2. Mission Overview of Sample Aircraft

<table>
<thead>
<tr>
<th>Phase</th>
<th>Time (min)</th>
<th>Distance (nm)</th>
<th>Fuel (kg)</th>
<th>% Block Fuel</th>
</tr>
</thead>
<tbody>
<tr>
<td>Taxi-out</td>
<td>3.0</td>
<td>0.0</td>
<td>9</td>
<td>2.6%</td>
</tr>
<tr>
<td>Take-off</td>
<td>1.3</td>
<td>0.0</td>
<td>21</td>
<td>4.8%</td>
</tr>
<tr>
<td>Climb</td>
<td>16.1</td>
<td>54.6</td>
<td>177</td>
<td>40.0%</td>
</tr>
<tr>
<td>Cruise</td>
<td>22.0</td>
<td>101.0</td>
<td>186</td>
<td>42.1%</td>
</tr>
<tr>
<td>Descent</td>
<td>9.9</td>
<td>44.4</td>
<td>32</td>
<td>7.2%</td>
</tr>
<tr>
<td>Approach &amp; Landing</td>
<td>2.0</td>
<td>0.0</td>
<td>14</td>
<td>3.5%</td>
</tr>
<tr>
<td>Taxi in</td>
<td>1.0</td>
<td>0.0</td>
<td>3</td>
<td>0.7%</td>
</tr>
<tr>
<td>Total</td>
<td>55.3</td>
<td>200.0</td>
<td>442</td>
<td>100.0%</td>
</tr>
</tbody>
</table>

Figure 7. Mission Profile Comparison of Sample Aircraft

Utilizing gravitational potential energy during descent by windmilling propellers produces energy that can be stored in the EES. On the experimental aircraft, four different propeller modes were tested: folded, feathered, transparency, and wind turbine. Folded propeller blades ideally produce zero drag during descent. Feathered blades rotate parallel to the direction of the airflow. These add to the drag of the aircraft. Transparency blades result in rotating propellers that don’t generate thrust or drag. Wind turbine blades add drag and energy to the EES. Results from these trade studies demonstrated that propellers should be folded during descent, and with a hybrid power generation system, transparency mode is most beneficial during normal operations. (Thauvin, et al., 2016)
Other results regarding power efficiency provided additional improvements in operation. The propeller efficiency could be increased by 49% if a hybrid electric system is utilized during taxi, removing the minimum speed constraint. Energy during taxi could be decreased by 90% if the taxi and descent phase could operate on purely electric power. Another consideration could be to use a single larger engine that has two times the power. This improves efficiency by 10.5%, but removes redundancies in the event of engine failure. Finally, the engine could be sized to perform the climbing phase, with that phase being the most energy demanding. (Thauvin, et al., 2016) Each of these considerations could be used to analyze and design future aircraft of various sizes and explore the balance of the two power sources.

Other considerations for different types of batteries that could prove more energy dense in future years have also been studied. Lithium ion batteries currently have a mass specific energy content of Wh/kg, with a potential of 250 Wh/kg. Other variations of lithium batteries could produce higher specific energies. Research is being conducted on Lithium-Sulfur (Li-S) and Lithium-Oxygen (Li-O) batteries. Additionally, Zinc-Air (Zn-air) also have higher theoretical specific densities. Their specific densities are summarized in Table 3, and an illustration of their performance is present in Figure 8. (Hepperle) These batteries could potentially provide an AV-gas fuel equivalent of 3800 Wh/kg, and power an electric aircraft. However, current technology is still limited. Hence,
the hybrid aircraft could provide an alternative aircraft which combines current battery technology with reduced fuel and emissions.

Table 3. Summary of Theoretical Specific Energy

<table>
<thead>
<tr>
<th>System</th>
<th>theoretical specific energy</th>
<th>expected in 2025</th>
</tr>
</thead>
<tbody>
<tr>
<td>Li-Ion (2012)</td>
<td>390 Wh/kg</td>
<td>250 Wh/kg</td>
</tr>
<tr>
<td>Zn-air</td>
<td>1090 Wh/kg</td>
<td>400-500 Wh/kg</td>
</tr>
<tr>
<td>Li-S</td>
<td>2570 Wh/kg</td>
<td>500-1250 Wh/kg</td>
</tr>
<tr>
<td>Li-O2</td>
<td>3500 Wh/kg</td>
<td>800-1750 Wh/kg</td>
</tr>
</tbody>
</table>

Figure 8. Current and expected battery technology

1.4. Project Objective

The purpose and objective of this project is to design a hybrid powered aircraft with the following mission specifications. The payload will consist of a FAA maximum of 124 passengers and 2 crew members along with their luggage. The aircraft will have a maximum range of 1720 nautical miles at maximum payload and a cruise speed of 500 kts. The required takeoff field length will be 5,500 ft, and the required landing distance is 3,960 ft. The maximum cruise altitude will be 40,000 ft. With the use of alternative power, provided by both Lithium ion batteries and fuel, the project will focus on design trade-offs to power the short-range, narrow-body aircraft that will reduce fuel usage, and provide a cheaper, cleaner, more energy-efficient hybrid commercial vehicle.
1.5. Mission Specifications

1.5.1. Mission Specifications

The following consist of mission requirements for the HTA:

- Payload capacity: All Economy with 96 at 6 abreast; FAA limit: 124
- Number of crew members required: 2 pilots, 2 cabin attendants
- Range: 1720 nm
- Cruise speed and Mach number: 500 kts (M=0.75)
- Cruise altitude: 40,000 ft
- Take-off field length: 5,500 ft
- Landing field length: 3,960 ft
- Approach speed: 128 knots
- Noise requirements: $\leq 97.5$ EPNdb

1.5.2. Mission Profile

Figure 9. Mission Profile
1.6. Methodology

This hybrid aircraft will be designed and sized through a combination of techniques that will account for both the fuel and batteries. Initially, Roskam’s approach using an estimated take-off weight and fuel fractions will be used to address the changing fuel weight during the various phases of the flight. The battery weight will be constant and will be sized using the work of Riboldi and Gualdoni. Trade studies will be performed to assess the best usage of fuel and batteries during each flight phase.

Once the weight has been sized, performance sizing and verification will be performed, using the smallest size engine with the lowest weight. Then the design of the fuselage will take place. This will be based on the 737-100 series aircraft, but modifications may be made to suit the payload and change in power supply. From there, the wing will be sized. Again, this will be based on the 737-100 aircraft, but changes may be made if parametric studies support changes in sweep angle, wing thickness, or taper ratio. High lift systems will be applied if they are beneficial. Analysis on the empinage design will be similar to the analysis of the wing. The last feature of the aircraft to design will be the landing gear that promotes stability and control and maintains the appropriate weight.

Drag analysis will additionally be performed to estimate the drag of the entire aircraft and calculate a L:D ratio. If the aircraft, after all additions and design changes, is within 0.5% of its initial weight, small changes will be made to reduce or resize. However, if the weight is not within 0.5%, the aircraft sizing will need to be iterated until it is within 0.5% of the initial weight.

1.7. Comparative Study of Airplanes with Similar Mission Performance

1.7.1. Comparison of Weights, Performance, and Geometries of Similar Airplanes

Table 4. Comparison of Aircraft Weights
### Table 5. Comparison of Aircraft Performance (Assuming standard conditions at sea level)

<table>
<thead>
<tr>
<th></th>
<th>Boeing 737-100</th>
<th>Boeing 737-200</th>
<th>Airbus A320NEO</th>
<th>Embraer ERJ-170-100</th>
<th>ARJ21-700STD</th>
<th>Bombardier CRJ200-ER</th>
<th>SUGAR Volt 765-096-RevA</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Max Cruise Speed</strong></td>
<td>0.82</td>
<td>0.82</td>
<td>0.82</td>
<td>0.82</td>
<td>0.82</td>
<td>0.85</td>
<td>0.7</td>
</tr>
<tr>
<td>(Mach number)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Cruise Speed</strong></td>
<td>580</td>
<td>580</td>
<td>598</td>
<td>529</td>
<td>518</td>
<td>534</td>
<td>537</td>
</tr>
<tr>
<td>(mph)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Takeoff Field</strong></td>
<td>5,499</td>
<td>5,499</td>
<td>6,857</td>
<td>4,866</td>
<td>5,578</td>
<td>5,800</td>
<td>8,180</td>
</tr>
<tr>
<td>Length (MTOW) (ft)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Landing Field</strong></td>
<td>3,960</td>
<td>3,960</td>
<td>5,020</td>
<td>4,029</td>
<td>5,086</td>
<td>4,850</td>
<td>--</td>
</tr>
<tr>
<td>Length (ft)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Service Ceiling</strong></td>
<td>35,000</td>
<td>37,000</td>
<td>19,500</td>
<td>41,000</td>
<td>20,340</td>
<td>41,000</td>
<td>--</td>
</tr>
<tr>
<td>(ft)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
## Design of a Hybrid Transport Aircraft

### Table 6. Comparison of Aircraft Geometries

<table>
<thead>
<tr>
<th>Cruising Altitude (ft)</th>
<th>Range (nm)</th>
<th>Source(s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>23,500</td>
<td>1,720</td>
<td>(Boeing Commercial Airplanes, 2013)</td>
</tr>
<tr>
<td>23,500</td>
<td>2,645</td>
<td>(Boeing Commercial Airplanes, 2013)</td>
</tr>
<tr>
<td>37,000</td>
<td>3,078 (3,700 with sharklets)</td>
<td>(Airbus Commercial Aircraft, n.d.) (Jane's All the World's Aircraft, 2013)</td>
</tr>
<tr>
<td>41,000</td>
<td>1,800</td>
<td>(Jane's All the World's Aircraft, 2013)</td>
</tr>
<tr>
<td>35,000</td>
<td>1,200</td>
<td>(Jane's All the World's Aircraft, 2013)</td>
</tr>
<tr>
<td>42,000</td>
<td>1,893</td>
<td>(Bradley &amp; Droney, Subsonic Ultra Green Aircraft Research: Phase II-Volume II-Hybrid Electric Design Exploration, 2015)</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Boeing 737-100</th>
<th>Boeing 737-200</th>
<th>Airbus A320NEO</th>
<th>Embraer ERJ-170-100</th>
<th>ARJ21-700STD</th>
<th>CRJ200-ER</th>
<th>SUGAR Volt 765-096-RevA</th>
</tr>
</thead>
<tbody>
<tr>
<td>Overall length (ft)</td>
<td>94 ft</td>
<td>100 ft 2 in</td>
<td>123 ft 3.25 in</td>
<td>98 ft 1.25 in</td>
<td>109 ft 9.5 in</td>
<td>87 ft 10 in</td>
</tr>
<tr>
<td>Height (ft) at max Woe</td>
<td>37 ft 2 in</td>
<td>37 ft 3 in</td>
<td>38 ft 7 in</td>
<td>31 ft 11.25 in</td>
<td>27 ft 8.25 in</td>
<td>20 ft 5 in</td>
</tr>
<tr>
<td>Wing Area (ft²)</td>
<td>1098</td>
<td>1098</td>
<td>1317.5</td>
<td>782.8</td>
<td>859.6</td>
<td>587.1</td>
</tr>
<tr>
<td>Wing span</td>
<td>93 ft</td>
<td>93 ft</td>
<td>117 ft 5.5 (including sharklets)</td>
<td>85 ft 3.5 in</td>
<td>89 ft 6.5 in</td>
<td>69 ft 7 in</td>
</tr>
<tr>
<td>Cabin width</td>
<td>12 ft 4 in</td>
<td>12 ft 4 in</td>
<td>12 ft 1.75 in</td>
<td>8 ft 11.75 in</td>
<td>10 ft 3.75 in</td>
<td>8 ft 5 in</td>
</tr>
<tr>
<td>Fuselage width</td>
<td>12 ft 4 in</td>
<td>12 ft 4 in</td>
<td>12 ft 11.5 in</td>
<td>9 ft 11 in</td>
<td>8 ft 10 in</td>
<td>148.7</td>
</tr>
<tr>
<td>Fuselage length</td>
<td>90 ft 7 in</td>
<td>96 ft 11 in</td>
<td>123 ft 3 in</td>
<td>98 ft 1 in</td>
<td>99 ft 10 in</td>
<td>80 ft</td>
</tr>
<tr>
<td>Total Bulk Cargo (ft³)</td>
<td>650</td>
<td>875</td>
<td>403</td>
<td>515</td>
<td>711.4</td>
<td>318</td>
</tr>
</tbody>
</table>
1.7.2. Configuration Comparison of Similar Airplanes

*Figure 10.* Boeing 737-100 aircraft configurations (*Boeing Commercial Airplanes, 2013*)
Figure 11. Boeing 737-200 aircraft configuration. *(Boeing Commercial Airplanes, 2013)*
Figure 12. Airbus A320NEO aircraft configuration (Airbus Commercial Aircraft, n.d.)
Figure 13. Embraer ERJ-170-100 aircraft configuration (Jane's All the World's Aircraft, 2013)
Figure 14. CRJ200-ER aircraft configuration (Canadair Regional Jet, 2016)
1.8. Discussion

The configuration designs of the 737-100, 737-200, A320NEO, ERJ-170-100, CRJ200-ER, and SUGAR Volt 765-096 RevA demonstrate common characteristics of transport aircraft, as observed by Figure 10. Boeing 737-100 aircraft configurations Figure 15. SUGAR Volt 765-096 RevA aircraft configuration Aircraft configurations allow for many architectural options. Among these options are the wing sweep, wing style, wing position, propulsion system integration, and other added aerodynamic features. Each of these features contributes to the aircraft’s ideal design for performance, design for handling qualities, and meeting FAA requirements on safety, ability, and noise.
The aircraft configurations display wings that have been aft swept. The sweep angle depends on the speed of the aircraft. Subsonic speeds allow for small sweep angles. Transonic speeds delay increase in drag, due to higher Mach numbers, by sweeping the wings 30° -35°. Supersonic aircraft have sweep angles of 45° -70°. (Mason, 2006) The sampled aircraft have small sweep angles, with the Boeing 737-100 having a sweep angle of 25° and the SUGAR Volt 765-096 RevA having sweep angle of 12.52°. (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase II- Volume II- Hybrid Electric Design Exploration, 2015)

Each of the aircraft also features a low wing, excluding the SUGAR Volt 765-096 RevA. This typically provides efficient use of the fuselage cargo space and easy retraction of landing gear. This characteristic also allows for better maneuverability and smooth landing. The SUGAR Volt aircraft had additional systems to integrate involving batteries, which changed the wing configuration. It features a high wing Truss Braced Wing (TBW), which is predicted to save fuel consumption by 5-10% compared with conventional low wings. (NASA, 2014)

For the propulsion integration, the 737-100, 737-200, A320NEO, and ERJ-170-100 have two pod mounted engines suspended below the wings. This meets civil aircraft requirements of having more than one engine in the event that one becomes inoperable. The airplane must be able to complete take-off in the event of one engine failure or only 50% power with 2 engines. (Obert, 2009) The CRJ200-ER has two pod mounted engines fitted to the rear of the fuselage. These two locations, below the wings and on the aft of the fuselage, are common configurations for jet transport engines. Boeing has demonstrated that suspended engines don’t incur a drag penalty. (Obert, 2009) This structural design of placing engines in nacelles suspended below the wings has several advantages. Among these are load relief, with the engines opposing the lift force; easy engine access for maintenance and repairs; and safety since the engines are further
Design of a Hybrid Transport Aircraft

from the passengers. (Obert, 2009) The SUGAR Volt 765-096 RevA has two engines suspended below the wings along with two battery pods suspended below the wings. This configuration allows for easy charge or exchange of batteries for quick transitions between flights. Additionally, it balances the load of the battery pod weight. (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase II- Volume II- Hybrid Electric Design Exploration, 2015)

The tail and fin configurations also differ for the CRJ200-ER and SUGAR Volt 765-096 RevA aircraft compared to the 737-100, 737-200, A230NEO, and ERJ-170-100. The CRJ200-ER and SUGAR Volt 765-096 RevA feature a T-tail; while the other aircraft feature fuselage mounted conventional tails. Along with the fuselage mounted tails, the aircraft also have tail plane mounted fins. These different configurations provide different aircraft control. A T-tail, which is an empennage configuration where the tailplane is attached to the top of the fin, typically accompanies fuselage mounted engines. With a T-tail arrangement, there is a risk for the horizontal tail becoming engrossed by the wake of the wing at high angles of attack. (Mason, 2006) This is likely why the more common configuration is the conventional fuselage mounted tail.

The Airbus A320NEO also has added sharklets to their wings. The Sharklet wingtip devices are 7.9 feet wingtips that are standard on all NEO aircraft. They reduce fuel burn by approximately 4%, and reduce annual CO₂ emissions by approximately 9,000 tons per aircraft. (Airbus Commerical Aircraft, n.d.) This added feature could be added to other A320 aircraft as well as other aircraft models.

In summary, choosing various configurations can improve safety, increase performance, and, with added features, reduce fuel consumption. The four common aircraft configurations illustrate that the low swept wing, with two pod mounted engines suspended from the wing and
tail plane mounted fins meets the requirements of short to mid-range flights by providing greater structural efficiency, less weight, and satisfactory stall characteristics. New advances in design also demonstrate the benefits of the high wing TBW, as well as integration of batteries for hybrid and electric vehicles.

2. Configuration Selection

2.1. Overall Configuration

The HTA will have a conventional configuration due to the expansive database of conventional aircraft and design modeled after the Boeing 737 aircraft. From here, the configurations of the fuselage, engine, wing, empennage, and landing gear will be presented with short discussions on the rationale for selections.

2.1.1. Fuselage Configuration

The conventional fuselage will be designed to hold two cockpit crew members, 96 passengers at 6 abreast, and 2 flight attendants. Since it is designed for a short to medium range flight, an approximate value of 175 lbs per person and 30 lbs of luggage per person will be chosen for both the crew and the passengers. This totals 20,500 lbs, with 3,000 lbs being designated for cargo. This cargo will be stored below the cabin floor in cargo containers that have a capacity of 80 ft$^3$. According to Roskam, typical luggage density is 12.5 lb/ft$^3$ (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017). Therefore, 240 ft$^3$ of baggage volume will be required for bagge storage in 3 containers.

With 6 abreast seating, there will be 16 rows, which maintains a shorter fuselage. Although 5 abreast seating may be more comfortable for passengers, this would add an additional 4 rows of seating, which increases the fuselage length and the aircraft weight. The seat pitch for tourist/economy/coach passengers ranges from 34-36 inches, and could be 30-32
inches for high density seating arrangements. (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017) The hybrid aircraft will have a 34 in. seat pitch, and 18 in. aisles to provide a comfortable mode of transportation. Further design layouts of the fuselage will be presented in the fuselage design section.

2.1.2. Engine Configuration

The hybrid aircraft will have two engines suspended below the wings. Two engines provides for redundancy in the event of a single engine failure or 50% failure of both engines. The engines will be stored in nacelles, and two battery pods will also be stored in nacelles attached under the wing. Having the batteries outside of the aircraft allows for easier access when charging and easier exchange of the batteries for quick transitions between flights. 737-100 model airplanes used JT8D turbofan engines. For the hybrid airplane, the design could include two hFan engines with 1,380 hp and thrust equivalent to 21,000 lbf of Boeing thrust.

Although buried engines produce less drag, having the engines stored in pods outside the fuselage also provides the ability to easily repair or exchange as technology improves. This configuration also provides optimal engine operation, in-flight wing bending relief, and engine stability. Mounting the engines above the wings also has advantages, including less debris ingestion and lower noise levels. However, this is not beneficial aerodynamically, and will not be considered.

2.1.3. Wing Configuration

For the wing configuration, the airplane will have a low Cantilever wing. Low wings are ideal for transport vehicles due to the increased safety of the aircraft, increased cargo volume, and ease of maneuverability. High wings are generally best for short take-offs and landings, but offer very little cargo space. Mid wings provide the least drag during flight due to their design,
with the wings being continuous with the fuselage. This design would not be ideal for the hybrid transport due to the fact that the fuselage volume is reduced to accommodate the wings.

The wings will be swept aft with the characteristics of the wings, including sweep angle, aspect ratio, thickness ratio, and other features, being discussed and explored in detail later.

2.1.4. Empennage Configuration

The empennage consists of the stabilizing tail and added features on the aircraft. The configuration for the hybrid transport aircraft will be a single vertical tail mounted to the fuselage. Mounting to the fuselage provides adequate support. Since the tails are designed to help recover from stalling and spinning, the vertical configuration is more reliable than the T-tail configuration, which may be affected by the air flow, potentially causing deep stall.

2.1.5. Landing Gear Configuration

Since the hybrid vehicle is a narrow-body vehicle, the bicycle landing gear can easily be used. This does limit the takeoff and landing locations to level ground since the landing gear will not be able to rotate. With retractable landing gear mounted on the fuselage, the drag during flight will not be increased, and the gears are well-supported. Having the landing gear mounted to the nose of the fuselage provides more stability than aft wheel mounted landing gear. However, this position increases the weight. Since weight is such a large factor in the design, due to the already heavy batteries, mounting the retractable landing gear to the aft of the wheel would be beneficial.

The remaining aspects of the configuration, including number of main fear struts, number of tires, and the retraction kinematics will be discussed in detail when the landing gear is sized.

In summary the overall configuration of the aircraft will be a conventional fuselage designed to comfortably hold 96 passengers and their luggage, The aircraft will be supported by
conventional low wings with 2 engines mounted below the wings and 2 batteries mounted below the wings. The vertical tail will provide the stability during flight, and the tricycle aft wheel retractable landing gear will provide adequate support with minimal weight for landing.

3. **Weight Sizing and Weight Sensitivities**

Using data for takeoff weights and empty weights of similar aircraft, a plot of gross takeoff weight vs. empty weight will be analyzed. The axes will be scaled logarithmically to create a log-log plot of the weight data. The regression coefficients will be calculated to determine the weight required for takeoff and for the overall mission. A sensitivity study will be performed using the AAA Program.

3.1 **Data Base for Takeoff Weights and Empty Weights of Similar Airplanes**

The following data in Table 7 provide takeoff weights and empty weights for comparison to the HTA. Roskam additionally provides transport jet weights for comparison. The takeoff weights and empty weights can be related linearly when graphing the log of the takeoff weight and the log of the empty weight. The plot of this relationship is illustrated in Figure 16.

*Table 7. Aircraft Takeoff and Empty Weights*

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Takeoff Weight (lbs)</th>
<th>Empty Weight (lbs)</th>
<th>Source</th>
</tr>
</thead>
<tbody>
<tr>
<td>Balanced 765-095 (1,380 Hp)</td>
<td>139,700</td>
<td>83,800</td>
<td>(Bradley &amp; Droney, Subsonic Ultra Green Aircraft Research: Phase II- Volume II-Hybrid Electric Design Exploration, 2015)</td>
</tr>
<tr>
<td>Core Shutdown 765-095</td>
<td>190,000</td>
<td>102,600</td>
<td>(Bradley &amp; Droney, Subsonic Ultra Green Aircraft Research: Phase II- Volume II-Hybrid Electric Design Exploration, 2015)</td>
</tr>
<tr>
<td>Boeing 737-100</td>
<td>97,000</td>
<td>58,600</td>
<td>(Boeing Commercial Airplanes, 2013)</td>
</tr>
<tr>
<td>Boeing 737-200</td>
<td>100,000</td>
<td>59,900</td>
<td>(Boeing Commercial Airplanes, 2013)</td>
</tr>
<tr>
<td>Airbus A320NEO</td>
<td>174,165</td>
<td>92,814</td>
<td>(Jane's All the World's Aircraft, 2013)</td>
</tr>
<tr>
<td>Embraer ERJ-170-100</td>
<td>79,344</td>
<td>48,733</td>
<td>(Jane's All the World's Aircraft, 2013)</td>
</tr>
<tr>
<td>ARJ21-700STD</td>
<td>89,287</td>
<td>55,016</td>
<td>(Jane's All the World's Aircraft, 2013)</td>
</tr>
<tr>
<td>CRJ200- ER</td>
<td>51,000</td>
<td>30,500</td>
<td>(Jane's All the World's Aircraft, 2013)</td>
</tr>
</tbody>
</table>
From the plot, the regression line constants A and B can be determined, since the log of the weights has the following relationship:

$$\log(W_{to}) = A + B \log(W_\varepsilon)$$

Thus, $A = 0.1776$ and $B = 0.9195$, as demonstrated by the regression equation in the plot. In Part 1 of Roskam’s Airplane Design Volumes, Roskam lists the regression line constants for various airplane types. These are included in Figure 17. The B values obtained in this analysis are similar, with B having an 11.44% margin of error. However, the A values are very different. This could be due to the type of aircraft in the database. 5 of the 12 aircraft included are hybrid vehicles. The remaining 7 are conventional fuel-powered aircraft. The A value for this database differs by -113.2%, with the transport jet A coefficient calculated as 0.0833 and the B value as 1.0383.

The other factor that could impact the weights could be the materials used for the primary and secondary structures, which would affect the takeoff weight and empty weight. Roskam’s database consists of Boeing 727, 737, 747, 757 and 767 aircraft, McDonnell Douglas DC 8, 9, and 10 aircraft, Airbus A300 and A310 aircraft, and several others. The Boeing aircraft were constructed from aluminum alloy, aluminum, and graphite composites. (Jane's All the World's
Aircraft, 2013) The Airbus aircraft were constructed from aluminum alloy, steel, and titanium. The SUGAR aircraft have incorporated metal-matrix composites, ceramics, high temperature polymer composites, thermoplastic composites, carbon, and nanocomposites. These will lighten the weight of the structure, landing gear, and engine nacelles. (Bradley, Droney, & Allen, 2015)

<table>
<thead>
<tr>
<th>Airplane Type</th>
<th>A</th>
<th>B</th>
<th>Airplane Type</th>
<th>A</th>
<th>B</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Homebuilts</td>
<td></td>
<td></td>
<td>8. Military Trainers</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Pers. fun and</td>
<td>0.3411</td>
<td>0.9519</td>
<td>Jets</td>
<td>0.6632</td>
<td>0.8640</td>
</tr>
<tr>
<td>transportation</td>
<td></td>
<td></td>
<td>Turboprops</td>
<td>-1.4041</td>
<td>1.4660</td>
</tr>
<tr>
<td>Scaled Fighters</td>
<td>0.5142</td>
<td>0.8654</td>
<td>Turboprops</td>
<td>0.1677</td>
<td>0.9978</td>
</tr>
<tr>
<td>Composites</td>
<td>0.8222</td>
<td>0.8050</td>
<td>without No. 2</td>
<td>0.5627</td>
<td>0.8761</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Piston/Props</td>
<td>0.5627</td>
<td>0.8761</td>
</tr>
<tr>
<td>Propeller Driven</td>
<td>-0.1440</td>
<td>1.1162</td>
<td>Jets (+ ext. load)</td>
<td>0.5091</td>
<td>0.9505</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Jets (clean)</td>
<td>0.1362</td>
<td>1.0116</td>
</tr>
<tr>
<td>3. Twin Engine</td>
<td></td>
<td></td>
<td>Turboprops (+ ext. load)</td>
<td>0.2705</td>
<td>0.9830</td>
</tr>
<tr>
<td>Propeller Driven</td>
<td>0.0966</td>
<td>1.0298</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Composites</td>
<td>0.1130</td>
<td>1.0403</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4. Agricultural</td>
<td></td>
<td></td>
<td>10. Mil. Patrol, Bomb and Transport</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>-0.4198</td>
<td>1.1546</td>
<td>Jets</td>
<td>-0.2099</td>
<td>1.1037</td>
</tr>
<tr>
<td>5. Business Jets</td>
<td>0.2678</td>
<td>0.9579</td>
<td>Turboprops</td>
<td>-0.4179</td>
<td>1.1446</td>
</tr>
<tr>
<td>6. Regional TBP</td>
<td>0.3174</td>
<td>0.9647</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>7. Transport Jets</td>
<td>0.0833</td>
<td>1.0183</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 2.15 Regression Line Constants A and B of Equation (2.16)

Figure 17. Regression Line Constants for Aircraft

For future calculations, both Roskam’s constants and the calculated constants will initially be used and comparisons in results will be made before determining the constant that will be used in the end.

3.2. Determination of Mission Weights

The following mission weights will be estimated: \( W_{pl} \), \( W_{to} \), \( W_f \), \( W_e \), and \( W_{bat} \) using comparisons to similar aircraft, calculations from Roskam and Riboldi, and iterating through the AAA program to verify and refine estimations.

Takeoff weight, \( W_{to} \), consists of the sum of the operating empty weight of the aircraft, \( W_{oe} \), with the weight of the fuel, \( W_f \), weight of the payload, \( W_{pl} \) and the added weight of the
\[ W_{to} = W_{oe} + W_f + W_{pl} + W_{bat} \] (10).

\[ W_{to} = W_{oe} + W_f + W_{pl} + W_{bat} \] (10)

Note that the operating empty weight consists of the sum of the empty weight, weight of the trapped fuel and oil, and the weight of the crew, as summarized in \( W_{oe} = W_e + W_{tfo} + W_{crew} \) (11).

\[ W_{oe} = W_e + W_{tfo} + W_{crew} \] (11)

For \( W_{to} = W_{oe} + W_f + W_{pl} + W_{bat} \) (10), the weights of each of the parts will need to be calculated. Although the payload weight is known, the remaining values will need to be calculated. The process for this will be conducted as follows.

Initially, a takeoff weight will be estimated using data from similar aircraft. From there, the weights of the fuel, and batteries will be calculated. Roskam’s fuel fraction method will be used to calculate the mission fuel weight; while Riboldi’s battery sizing method will be employed to estimate mission battery weight. Then, the operating empty weight will be calculated to find an allowable value of the empty weight. AAA will be used to iterate and find an empty weight within 0.5% of the tentative empty weight.

### 3.3. Manual Calculation of Mission Weights

#### 3.3.1. Payload Weight

The payload weight consists of the passengers and their luggage. Based on the desired configuration of the cabin, 96 passengers will be assumed to weigh 175 lbs each and carry 30 lbs of luggage. Thus the total payload weight is 19,680 lbs.

#### 3.3.2. Estimated Takeoff Weight Method 1

To estimate the takeoff weight, there are two methods. The first consists of examining a database of aircraft and comparing \( W_{to} \) to \( W_{pl} \). The database, displayed in Table 8, consists of turbojet transport aircraft and conceptual hybrid aircraft. Using the data, an average ratio of \( W_{pl} \)
to $W_{to}$ was calculated as 0.237. Since the payload weight is a known value, the takeoff weight can be estimated using the ratio. Using a 96 passenger payload yields a takeoff weight value of 83,006.38 lbs. If the payload is increased to hold 100 passengers, the takeoff weight increases to 86,464.98 lbs. Finally, if the payload is increased to 108 passengers, the takeoff weight estimate is 93,382.18 lbs.

*Table 8. Data for $W_{to}$ Comparison*

<table>
<thead>
<tr>
<th></th>
<th>737-100</th>
<th>737-200</th>
<th>A320NEO</th>
<th>ERJ-170-100</th>
<th>ARJ21-700STD</th>
<th>CRJ200-ER</th>
<th>SUGAR Volt 765-096-RevA</th>
<th>737-300</th>
<th>McDD DC9-80</th>
<th>A320</th>
</tr>
</thead>
<tbody>
<tr>
<td>$W_{to}$</td>
<td>97,000</td>
<td>100,000</td>
<td>174,165</td>
<td>79,344</td>
<td>89,287</td>
<td>51,000</td>
<td>150,000</td>
<td>135,000</td>
<td>140,000</td>
<td>145,000</td>
</tr>
<tr>
<td>$W_{pl}$</td>
<td>23,100</td>
<td>25,100</td>
<td>44,974</td>
<td>19,918</td>
<td>19,698</td>
<td>13,500</td>
<td>30,800</td>
<td>35,000</td>
<td>38,000</td>
<td>42,000</td>
</tr>
</tbody>
</table>

The second method to calculate takeoff weight involves knowledge of more values. Therefore, it will be discussed later. With an estimation of takeoff weight from the database, calculating the fuel weight will be the next step.

**3.3.3. Determination of Mission Fuel Weight**

To determine the mission fuel weight, $W_f$, the weight of the fuel used during the mission and the weight of the reserve fuel will need to be known. To calculate the weight of the fuel for the mission, Roskam’s fuel fraction method will be employed. This consists of breaking down the mission into phases and estimating the fuel for each phase. The phases for the mission are: Phase 1: Engine start and warm-up, Phase 2: Taxi, Phase 3: Takeoff, Phase 4: Climb, Phase 5 Cruise, Phase 6: Loiter, Phase 7: Descent, and Phase 8: Land, taxi, and shut-down. This process will be presented for the 96 passenger payload and considered for a fuel powered flight. From there, fuel can be reduced by using batteries to power various phases of flight and eliminating parts of the fuel fraction process. AAA will then be used to calculate weights for the aircraft if it had used all fuel, to compare, as well as using fuel for stages 1-4 vs. fuel for stages 1-3, and 5.
To calculate the mission fuel weight, the sum of the fuel used and the fuel in reserves will need to be known.

\[ W_f = W_{\text{used}} + W_{\text{res}} \]  

(12)

\( W_{\text{used}} \) is the fuel used during the mission; while \( W_{\text{res}} \) is the fuel reserves required for the mission. These reserves are specified by FAR regulations. For a fully fuel-powered aircraft, the reserves will be 25% of the fuel used. The reserve quantities will change depending on the fuel used during different mission phases. The fuel used will be calculated using the fuel fraction method. Each phase will have an estimated or calculated fuel fraction, which is the ratio of the ending weight of the phase to the beginning weight of the phase.

*Table 9. Suggested Fuel-Fractions for Mission Phases*

<table>
<thead>
<tr>
<th>Mission Phase No. (See Fig. 2.1)</th>
<th>Engine Start, Warm-up</th>
<th>Taxi</th>
<th>Take-off</th>
<th>Climb</th>
<th>Descent</th>
<th>Landing Taxi, Shutdown</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Homebuilt</td>
<td>0.998</td>
<td>0.998</td>
<td>0.998</td>
<td>0.995</td>
<td>0.995</td>
<td>0.995</td>
</tr>
<tr>
<td>2. Single Engine</td>
<td>0.995</td>
<td>0.997</td>
<td>0.998</td>
<td>0.992</td>
<td>0.993</td>
<td>0.993</td>
</tr>
<tr>
<td>3. Twin Engine</td>
<td>0.992</td>
<td>0.996</td>
<td>0.996</td>
<td>0.990</td>
<td>0.992</td>
<td>0.992</td>
</tr>
<tr>
<td>4. Agricultural</td>
<td>0.996</td>
<td>0.995</td>
<td>0.996</td>
<td>0.998</td>
<td>0.999</td>
<td>0.998</td>
</tr>
<tr>
<td>5. Business Jets</td>
<td>0.990</td>
<td>0.995</td>
<td>0.995</td>
<td>0.980</td>
<td>0.990</td>
<td>0.992</td>
</tr>
<tr>
<td>6. Regional TBP's</td>
<td>0.990</td>
<td>0.995</td>
<td>0.995</td>
<td>0.985</td>
<td>0.985</td>
<td>0.995</td>
</tr>
<tr>
<td>7. Transport Jets</td>
<td>0.990</td>
<td>0.990</td>
<td>0.995</td>
<td>0.980</td>
<td>0.990</td>
<td>0.992</td>
</tr>
<tr>
<td>8. Military</td>
<td>0.990</td>
<td>0.990</td>
<td>0.990</td>
<td>0.980</td>
<td>0.990</td>
<td>0.995</td>
</tr>
<tr>
<td>Trainers</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>9. Fighters</td>
<td>0.990</td>
<td>0.990</td>
<td>0.990</td>
<td>0.96-0.90</td>
<td>0.990</td>
<td>0.995</td>
</tr>
<tr>
<td>10. Mil. Patrol. Bomb, Transport</td>
<td>0.990</td>
<td>0.990</td>
<td>0.995</td>
<td>0.980</td>
<td>0.990</td>
<td>0.992</td>
</tr>
<tr>
<td>11. Flying Boats, Amphibious, Float Airplanes</td>
<td>0.992</td>
<td>0.990</td>
<td>0.996</td>
<td>0.985</td>
<td>0.990</td>
<td>0.990</td>
</tr>
<tr>
<td>12. Supersonic Cruise</td>
<td>0.990</td>
<td>0.995</td>
<td>0.995</td>
<td>0.92-0.87</td>
<td>0.985</td>
<td>0.992</td>
</tr>
</tbody>
</table>

As seen in Table 9, the fuel fractions for phases 1, 2, 3, 4, 7, and 8, have been estimated and published by Roskam. (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017) These values for transport jets will be used in the calculation of the fuel-powered aircraft. The phase 1 fuel fraction of 0.990 is the ratio of \( W_1 / W_{to} \). The phase 2 fuel fraction of 0.990 is the ratio of
$W_2/W_1$. 0.995 is the ratio of $W_3/W_2$, which is the phase 3 fuel fraction. 0.980 is the ratio of $W_4/W_3$; 0.990 is the ratio of $W_7/W_8$, and 0.992 is the ratio of $W_8/W_9$. The fuel fractions, $W_5/W_4$ and $W_6/W_5$, for phases 5 and 6, respectively, will be calculated.

\[
R_{cr} = \left( \frac{V}{c_j} \right) \left( \frac{L}{D} \right)_{cr} \ln \left( \frac{W_4}{W_5} \right)
\]

(13)

The velocity for cruise is known, along with the range. The lift to drag ratio and specific fuel consumption values can be found in Roskam’s part 1. For transport jets in cruise, L/D ranges from 15-24s and $c_j$ ranges from 0.5-0.9. The following values will be used to estimate $W_5/W_4$:

- \( V = 500 \) kts, \( c_j = 0.5 \), \( \frac{L}{D} = 22 \), and \( R_{cr} = 1656 \) nm. This yields $W_5/W_4 = 0.928$.

For the loiter phase, Breguet’s endurance equation, equation \( Eltr = \left( \frac{1}{c_{j_{ltr}}} \right) \left( \frac{L}{D} \right)_{ltr} \ln \left( \frac{W_5}{W_6} \right) \)

(14) below, for jet airplanes can be used (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017).

\[
E_{ltr} = \left( \frac{1}{c_{j_{ltr}}} \right) \left( \frac{L}{D} \right)_{ltr} \ln \left( \frac{W_5}{W_6} \right)
\]

(14)

\( E_{ltr} \) is the loiter time in hours. According to Roskam, \( c_{j_{ltr}} \) ranges from 0.4-0.6, and L/D for the loiter phase ranges from 14-18. Using \( E_{ltr} = 0.75 \) hours, \( c_{j_{ltr}} = 0.4 \), and \( \frac{L}{D} = 18 \),

$W_5/W_6 = 0.981$.

Now that the fuel fractions for each phase are known, the mission fuel fraction, $M_{ff}$, can be calculated using

\[
M_{ff} = \left( \frac{W_1}{W_{to}} \right) \prod_{i=1}^{7} \frac{W_{i+1}}{W_i}
\]

\[ M_{ff} = \left( \frac{w_i}{W_{to}} \right) \prod_{i=1}^{n} \frac{w_{i+1}}{w_i} \]  

(15)

Therefore, \( M_{ff} = 0.99 \cdot 0.99 \cdot 0.995 \cdot 0.98 \cdot 0.928 \cdot 0.981 \cdot 0.99 \cdot 0.992 = 0.854 \). Using the

\[ W_{fused} = (1 - M_{ff})W_{to} \]  

(16)

3.3.4. 

Roskam’s A and B values will be used since the calculated B value will not yield a real solution for the equation and hence not provide an estimated takeoff weight. D is the sum of the crew and passenger weight along with their cargo. Assuming 96 passengers and 4 cabin and  

\[ \log_{10} W_{to} = A + B \cdot \log_{10}(C \cdot W_{to} - D) \]  

(17)  

\( M_{res} \) is the reserve fuel fraction; \( M_{ff} \) is the mission fuel fraction; and \( M_{to} \) is the trapped fuel and oil fraction. \( M_{to} \) is assumed to be 0.005, and \( M_{res} \) is assumed to be 0.25. (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017) \( M_{ff} \) varies based on the amount of fuel,
Design of a Hybrid Transport Aircraft

required. This calculation will be reviewed in the following section. For 100% fuel use, $M_{ff} = 0.854$. With fuel use during stages 1-4, $M_{ff} = 0.956$. With fuel use during stages 1-3 and 5, $M_{ff} = 0.905$. Three different values of $C$ were calculated based on the three different fuel options. $C_{all\ fuel} = 0.813$, $C_{1-4} = 0.9396$, and $C_{1-3,5} = 0.876$. These three different values provided

t

d

3.3.5. Determining Allowable Value for Empty Weight

To determine an allowable value for empty weight, for a fuel-powered aircraft, first a tentative value for the operating empty weight is calculated, using the following equation:

$$W_{o\text{tent}} = W_{o\text{guess}} - W_f - W_{pl}$$  \hspace{1cm} (19)

Since the $W_{o\text{guess}}$, $W_f$ and $W_{pl}$ are known, where $W_{o\text{guess}} = 83,006.38$ lbs, $W_f = 45,111.6$ lbs, and $W_{pl} = 19,680$ lbs, then $W_{o\text{tent}} = 48,214.8$ lbs. Then, a tentative value for the empty weight is calculated, using the following equation

$$W_{e\text{tent}} = W_{o\text{tent}} - W_{to\text{fo}} - W_{crew}$$  \hspace{1cm} (20)

Since this is a tentative estimate, $W_{to\text{fo}}$ will not be subtracted at this time. It can add as much as 0.5% of the takeoff weight, but can initially be neglected. (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017) With the weight of the crew and the tentative operating empty weight, $W_{e\text{tent}} = 48,214.8 - 820 = 47,394.8$ lbs. This tentative weight is then
compared to a calculated weight. If these values are not within 0.5%, adjustments are made to the takeoff weight guess and the process is iterated.

Figure 18 illustrates the trend of takeoff weights and empty weights for transport jets. The regression line for the data has the following equation:

$$W_e = invlog_{10}((log_{10} W_{to} - A)/B))$$

Since A and B are known, along with the guessed takeoff weight, $W_e$ is calculated as 47,394.8 lbs. Since these two weights are close, but not within 0.5% of each other, adjustments will be made to the takeoff weight guess and the empty weight will be recalculated. This process will be completed using AAA.
3.3.6.

To determine mission battery weight, the method of Riboldi and Gualdoni will be used.
specific power and specific energy values are values for lithium ion batteries by the year 2025, and will likely increase further with advancements in batteries. (Hepperle)

Since the climb, cruise, and loiter phases are the longest phases of the flight, the power and energy required for these phases will be taken into account. To calculate the power and energy required for each phase, the following calculations are performed:

\[
S = 1,341 \text{ ft}^2
\]

From there, the power for each phase can be calculated, using \( S = 1,341 \text{ ft}^2 \) and the velocity at cruise to be 575 mph. The power at loiter is 405.34 watts, and the power at cruise is 736.42 watts. The energy required for loiter is 1,094,411.5 joules, and the energy required for cruise is 8,779,440.79 joules.

### 3.4. AAA Calculation of Mission Weights

The following figures summarize the output parameters generated by AAA given the input parameters of the mission. The summary of data in the first two figures is for a twin turbojet engine fuel-powered aircraft. The manual calculations and AAA calculations proved to be very similar for the fuel-powered aircraft, with the AAA output weights being lower. AAA calculated the design point takeoff weight to be 71,947.6 lbs and the empty weight to be 39,775.9 lbs. The manual calculation produced an estimate of 77,602.6 lbs for the takeoff weight and 45,439.32 lbs for the empty weight. For an all fuel-aircraft, AAA calculated 11,554 lbs of fuel,
including 25% reserves; while manual calculations found 15,111.6 lbs of fuel necessary, including the 25% reserve fuel.

**Figure 19.** AAA output with all fuel aircraft with 2 turbojet engines

<table>
<thead>
<tr>
<th>Mission Profile</th>
<th>$W_{\text{deg}}$ lb</th>
<th>$\Delta W_{\text{fuel}}$ lb</th>
<th>$W_{\text{f}}$ lb</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 Warmup</td>
<td>71047.6</td>
<td>119.5</td>
<td>11554.0</td>
</tr>
<tr>
<td>2 Taxi</td>
<td>71228.1</td>
<td>712.3</td>
<td>10834.6</td>
</tr>
<tr>
<td>3 Take-off</td>
<td>70515.8</td>
<td>352.6</td>
<td>10127.2</td>
</tr>
<tr>
<td>4 Climb</td>
<td>70163.3</td>
<td>166.2</td>
<td>9769.7</td>
</tr>
<tr>
<td>5 Cruise</td>
<td>69950.9</td>
<td>5078.8</td>
<td>9803.5</td>
</tr>
<tr>
<td>6 Loler</td>
<td>64921.6</td>
<td>1073.1</td>
<td>4528.0</td>
</tr>
<tr>
<td>7 Descend</td>
<td>63848.5</td>
<td>938.5</td>
<td>3455.0</td>
</tr>
<tr>
<td>8 Land-Taxi</td>
<td>63219.8</td>
<td>505.7</td>
<td>2816.5</td>
</tr>
</tbody>
</table>

**Figure 20.** $W_e$ vs. $W_{10}$ of 2 turbojet engine fuel-powered aircraft with ideal design point of $W_{10}=71,635.9$ lbs and $W_e=39,775.9$ lbs
3.5. Takeoff Weight Sensitivities

3.5.1. Manual Calculations of Takeoff Weight Sensitivities

Aircraft takeoff weight depends on many factors. These include the payload weight, empty weight, range, endurance, lift to drag ratio, and specific fuel consumption. (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017) Sensitivity studies will be performed on these to determine which parameters impact the design. Using Roskam’s correlation coefficients of $A=0.083$ and $B=1.038$, the calculated $C$ values, the combined crew and passenger payload weight, $D=20,500 \text{ lbs}$, and the force, $F=250,337.6 \text{ lbs}$, sensitivities for the takeoff weight to: payload, empty weight, range, endurance, cruise speed, $L/D$, and specific fuel consumption will be calculated for the HTA. Additionally, the airplane growth factor due to payload will be calculated. To illustrate the concept, calculations will be completed for the fuel-powered aircraft, initially, with sensitivities also being calculated for various amounts of fuel use. This includes using fuel in stages 1-4, 1-3 and 5, and sensitivities for an electric aircraft.

In order to calculate the sensitivities, the takeoff weight needs to be established. For the purposes of these sensitivities, the takeoff weight used will be the manually calculated takeoff weight of 77,602.6 lbs. Using equation $\log(W_{to})= A + B \log(W_{e})$ (9) and taking the partial derivative of the takeoff weight with respect to the empty weight produces:

$$\frac{\partial W_{to}}{\partial W_{e}} = B W_{to} \left[ \text{invlog} \left( \frac{\log(W_{to}) - A}{B} \right) \right]^{-1}$$

Substituting the values gives: $\frac{\partial W_{to}}{\partial W_{e}} = (1.038)(77,602.6) \left[ \text{invlog} \left( \frac{\log(77,602.6) - 0.083}{1.038} \right) \right]^{-1} = 1.89$. As a result, for a fuel-powered aircraft, the takeoff weight must be increased by 1.89 lbs for each pound of increase in the empty weight. 1.89 is the growth factor due to empty weight for the fuel-powered aircraft. If the aircraft used fuel during phases 1-4, the growth factor would be
1.866, and it would remain the same for fuel use in phases 1-3 and 5. If the aircraft were all electric, the growth factor would be 2.22.

To compute the sensitivities of the takeoff weight to the payload weight, the partial derivative is:

\[
\frac{\partial W_{to}}{\partial W_{pl}} = B(W_{to}) \left\{ D - C(1 - B)W_{to} \right\}^{-1}
\]  
(23)

Substituting the remaining values produces \( \frac{\partial W_{to}}{\partial W_{pl}} = 1.038(77,602.6)(20,500 - 0.813(1 - 1.038)77,602.6)^{-1} = 3.5 \). Therefore, if the payload increases by 1 lb, the takeoff weight increases by 3.5 lbs for the fuel powered aircraft. When fuel is used in stages 1-4, the airplane growth factor due to payload is 2.5; with fuel use in stages 1-3, and 5, it’s 3.5; and for an electric airplane, the airplane growth factor due to payload is 2.2.

In order to calculate the sensitivity of takeoff weight to range, endurance, speed, specific fuel consumption, and lift to drag ratio, a generic case will be considered. The variable \( y \) will represent any parameter, excluding payload. Thus, the following equation represents the takeoff weight sensitivity to a parameter \( y \):

\[
\frac{\partial W_{to}}{\partial y} = \left\{ C W_{to} (1 - B) - D \right\}^{-1} B W_{to} \left( \frac{\partial C}{\partial y} \right)
\]  
(24)

Using Breguet’s equations and

\[
F = -B W_{to}^2 \left( C W_{to} (1 - B) - D \right)^{-1} (1 + M_{res}) M_{ff}
\]  
(25)

Breguet’s partials can be derived. They are available in Table 10, below. Using the manually calculated values, \( F=250,337.6 \) lbs.
Substituting in range and endurance for \( y \), respectively, yields

\[
\frac{\partial W_{to}}{\partial R} = F_c \left( V \left( \frac{L}{D} \right) \right)^{-1} = 16.7 \text{ lbs/nm} \quad \text{and} \quad \frac{\partial W_{to}}{\partial E} = F_c \left( \frac{L}{D} \right)^{-1} = 7,699.7 \text{ lbs/hr.}
\]

In summary, if the range of the fuel aircraft is decreased by 1 nm, the takeoff weight decreases by 16.7 lbs. If the endurance is increased by 1 hour, the takeoff weight is increased by 5,563 lbs.

Continuing to use Breguet’s partials in Table 10, the sensitivities of takeoff weight to cruise speed will be calculated. Assuming a cruise speed of 500 kts or \( M=0.75 \), \( L/D=22 \), cruise range of 1656 nm, and specific fuel consumption of 0.5, \( \frac{\partial W_{to}}{\partial V} = -37.7 \text{ lbs/kt.} \) Therefore, if the cruise speed could be increased without changing any parameters, the gross takeoff weight would decrease. Despite these results, this sensitivity result is not practical. An increase in cruise speed results in a decrease in the lift coefficient and thus a decrease in the lift to drag ratio. The specific fuel consumption will also change. (Roskam, Airplane Design Part 1: Preliminary
Sizing of Airplanes, 2017) All of these factors change the sensitivity of takeoff weight to cruise speed.

Finally, sensitivities of takeoff weight to L/D and to specific fuel consumption will be calculated during cruise and loiter. For the cruise phase, a cruise speed of 500 kts will be assumed, along with L/D ratio of 22, \( c_j = 0.5 \), and a force of 250,337.6 lbs. For the loiter phase, the lift to drag ratio will be 18, \( c_j = 0.4 \), and F=250,337.6 lbs. Using Breguet’s partials, the sensitivity of takeoff weight to L/D for range is -856.3; the sensitivity of takeoff weight to L/D for endurance is -154.5; the sensitivity of takeoff weight to \( c_j \) for range is 37,679.3; and the sensitivity of takeoff weight to \( c_j \) for endurance is 6,953.8. If L/D were increased, for both cruise and loiter, the takeoff weight would decrease. For cruise, if L/D were increased by 1, the takeoff weight would decrease by 856.3 lbs. Similarly. For loiter, if L/D were increased by 1, the takeoff weight would decrease by 154.5 lbs. For the specific fuel consumption, if \( c_j \) increases, the takeoff weight increases. For example, for the cruise phase, if \( c_j \) was incorrectly assumed to be 0.5 instead of 0.9, the takeoff weight would increase by \( 0.4 \times 37,679.3 \) or 15,071.7 lbs. Similarly, if if \( c_j \) was incorrectly assumed to be 0.4 instead of 0.6, during loiter, the takeoff weight would increase by \( 0.2 \times 6,953.8 \) or 1,390.8 lbs.

### 3.5.2. AAA Calculation of Takeoff Weight Sensitivities

The following data, displayed in Figure 21, summarizes AAA’s calculations of takeoff weight sensitivities for a fuel-powered aircraft with a takeoff weight of 71,947.6 lbs. According to AAA, if the payload were to increase by 1 lb, the takeoff weight would increase by 3.29 lbs. If the crew weight were to increase by 1 lbs, the takeoff weight would increase by 3.29 lbs. Finally, if the empty weight were to increase by 1 lb, the takeoff weight would increase by 2.04 lbs. For an electric aircraft, the takeoff weight increases by 2.17 lbs per pound increase of
payload. The takeoff weight increases by 2.17 lbs for every 1 pound increase in crew, and the if the empty weight increases by 1 pound, the takeoff weight increases by 1.86 lbs.

The sensitivities for the hybrid aircraft with fuel use in phases 1-4 or 1-3 and 5 have similar values with their values being between the fuel-powered aircraft and the electric aircraft. This holds true for both the manual calculations and the AAA calculations.

\[
\begin{array}{|c|c|c|c|}
\hline
\text{Input Parameters} & \text{Output Parameters} \\
\hline
W_{PL} & W_{TD} & W_{Crew} & W_{空} \\
116560.8 lb & 71947.6 lb & 760.8 lb & 39593.8 lb \\
\hline
M_{T1} & M_{T2} & M_{T3} & M_{T4} \\
8.8715 & 9.500 & 25.000 & 25.000 \\
\hline
\end{array}
\]

**Figure 21.** Takeoff weight sensitivities calculated by AAA for fuel-powered aircraft

**Figure 22.** Takeoff weight sensitivities calculated by AAA for electric aircraft

### 3.6. Trade Studies

Trade studies of takeoff weight versus L/D, payload, range, and specific fuel consumption will be performed. These trade studies are being performed on the HTA which will use fuel in stages 1-4. Additionally, a trade study of range vs. payload will be performed by AAA for the fuel-powered aircraft to compare.
3.7. Takeoff Weight vs. Critical Mission Parameters

3.7.1. Takeoff Weight vs. L/D

Figure 23. Takeoff Weight vs. L/D for Cruise

Figure 24. Takeoff Weight vs. L/D for Loiter
3.7.2. Takeoff Weight vs. Payload

![Graph of Takeoff Weight vs. Payload Weight](image)

*Figure 25. Takeoff Weight vs. Payload Weight (ranges from 80 passengers to 124 passengers)*

3.7.3. Takeoff Weight vs. Range

![Graph of Takeoff Weight vs. Range](image)

*Figure 26. Takeoff Weight vs. Range*
3.7.4. Takeoff Weight vs. Specific Fuel Consumption

**Figure 27.** Takeoff Weight vs. Specific Fuel Consumption during Cruise

**Figure 28.** Takeoff Weight vs. Specific Fuel Consumption during Loiter
3.8. AAA Calculation of Range vs. Payload

![Graph showing Range vs. Payload](image)

*Figure 29. Range vs. Payload*

Range vs. Payload with design point $W_{pl}=19,630.6$ lb and $R_{cr}=1,638.7$ nm for fuel-powered aircraft

3.9. Trade Study Conclusions

The estimated takeoff weight of 53,778.38 lbs, which is the sum of the manually calculated takeoff weight for the HTA and the batteries, assuming fuel is used during phases 1-4
of flight and lithium ion batteries supply the power for phases 5-8. This weight is represented in the trade study graphs and lies between the design points for the electric aircraft the the fuel-powered model.

The sensitivity studies demonstrated that the takeoff weight is strongly affected by the empty weight, the payload, specific fuel consumption, and lift to drag ratio. The aircraft is able to meet its mission specifications, based on the trade studies. From here constraint analyses will be performed, and the remaining systems will be designed.

4. Determination of Performance Constraints

Using FAR 25 requirements, data presented in Roskam’s Airplane Design volumes, and the HTA mission requirements, the performance constraints to which the HTA will be sized will be calculated and discussed. Calculations will be performed manually and with AAA. Then the results will be compared and analyzed. The performance constraints of stall speed, takeoff field length, landing field length, cruise speed, climb rate, time to climb to cruise, and maneuvering have an impact on the performance of the aircraft. Aircraft wing area, takeoff thrust (or takeoff power), maximum required lift coefficient for takeoff, and maximum required lift coefficient for landing include stall speed compose the airplane design parameters that are affected by the stall speed, takeoff field length, landing field length, cruise speed, climb rate, time to climb to cruise, and maneuvering.

Together, the results will determine a range of values for wing loading, W/S, thrust loading, T/W, and maximum lift coefficient, $C_{L,\text{max}}$. These values will be the result of meeting performance requirements, and specific values will be chose that allow for the lowest takeoff weight and minimize cost.
4.1. Manual Calculation of Performance Constraints

4.1.1. Stall Speed Sizing

The HTA will be designed to meet stall speed requirements, which, in turn, will define an allowable W/S for a given maximum lift coefficient. $C_{L_{max}}$ is influenced by the wing and airfoil design, flap type and size, and the center of gravity location (Roskam, Airplane Design Part 1:

\[
V_S = \left( \frac{2(W/S)}{\rho C_{L_{max}}} \right)^{1/2}
\]

(26)

e stall speeds during takeoff, cruise, and landing at different $M$ will be calculated. The 737-100 aircraft has a wing loading value of 433.33 kg/m$^2$. Assuming standard conditions on the ground and at altitude, $\rho = 1.225 \frac{kg}{m^3}$ and $\rho = 0.4135 \frac{kg}{m^3}$, respectively, $S = 102 m^2$ and varying $C_{L_{max}}$ values of 0.45, 0.35, and 0.3 at $M = 0.75, 0.74$, and 0.78, respectively, the stall speed during cruise with $M=0.74, C_{L_{max}} = 0.5$ at 40,000 ft is calculated as follows:

\[
V_S = \left( \frac{2(433.33 kg/m^2)}{0.4135 kg/m^3(0.5)} \right)^{1/2} = 125.8 kts
\]

The table below summarizes the stall speeds at various stages based on varying mach and lift coefficient values.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Stage</th>
<th>Cruise</th>
<th>Cruise</th>
<th>Cruise</th>
<th>Cruise</th>
<th>Takeoff</th>
<th>Landing</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_L$</td>
<td>0.45</td>
<td>0.5</td>
<td>0.3</td>
<td>0.58</td>
<td>1.75</td>
<td>2.73</td>
<td></td>
</tr>
<tr>
<td>$V_s$ (kts)</td>
<td>116.9</td>
<td>162.5</td>
<td>125.9</td>
<td>132.7</td>
<td>39.1</td>
<td>53.9</td>
<td></td>
</tr>
</tbody>
</table>

4.1.2. Takeoff Distance Sizing

According to Roskam, takeoff weight, takeoff speed, thrust to weight ratio at takeoff, the aerodynamic drag coefficient, and pilot technique all play a role in takeoff distance sizing.
(Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017). As a result, some parameters will be assumed to be fixed; while others will be changed to determine the thrust loading, T/W, for a given maximum lift coefficient and wing loading.

Since the field length required for takeoff, $S_{TOFL}$, is proportional to the wing loading, thrust to weight ratio, and the maximum lift coefficient at takeoff, the following equation is true:

$$ S_{TOFL} \propto \frac{(W/S)_{to}}{\sigma C_{\text{L,max}}(T/W)_{to}} = Top_{25} $$

(27)

Note that $Top_{25}$ is the takeoff parameter in lbs/ft$^2$, that meets FAR 25 requirements. Substituting in values for the density ratio, varying $C_{\text{L,max}}$ values, and varying wing loading values, various thrust loadings are calculated. As sample calculation with $\sigma = 0.861$ at 5000 ft elevation with standard conditions and $C_{\text{L,max}} = 1.2$ and $(W/S)_{to} = 40 \text{ lbs/ft}^2$ produces:

$$ \frac{40 \text{ lbs/ft}^2}{0.861(1.2)(217.3969 \text{ lbs/ft}^2)} = 0.1533 \text{ lb/lbf} = W/S_{to} $$

The following tables summarize varying lift coefficient values and wing loading values at various altitudes of takeoff and at various field lengths.

*Table 12. Takeoff Distance Sizing for $S_{TOFL} = 5000$ ft at 0 ft altitude*

<table>
<thead>
<tr>
<th>$C_{\text{L,max}_{to}}$</th>
<th>1.200</th>
<th>1.600</th>
<th>1.750</th>
<th>2.000</th>
<th>2.400</th>
</tr>
</thead>
<tbody>
<tr>
<td>$W/S_{to}$</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>40.000</td>
<td>0.250</td>
<td>0.188</td>
<td>0.171</td>
<td>0.150</td>
<td>0.125</td>
</tr>
<tr>
<td>60.000</td>
<td>0.375</td>
<td>0.281</td>
<td>0.257</td>
<td>0.225</td>
<td>0.188</td>
</tr>
<tr>
<td>80.000</td>
<td>0.500</td>
<td>0.375</td>
<td>0.343</td>
<td>0.300</td>
<td>0.250</td>
</tr>
<tr>
<td>100.000</td>
<td>0.625</td>
<td>0.469</td>
<td>0.429</td>
<td>0.375</td>
<td>0.313</td>
</tr>
<tr>
<td>120.000</td>
<td>0.750</td>
<td>0.563</td>
<td>0.514</td>
<td>0.450</td>
<td>0.375</td>
</tr>
<tr>
<td>140.000</td>
<td>0.875</td>
<td>0.656</td>
<td>0.600</td>
<td>0.525</td>
<td>0.438</td>
</tr>
<tr>
<td>160.000</td>
<td>1.000</td>
<td>0.750</td>
<td>0.686</td>
<td>0.600</td>
<td>0.500</td>
</tr>
</tbody>
</table>
Table 13. Takeoff Distance Sizing for $S_{TOFL} = 6000$ ft at 0 ft altitude

<table>
<thead>
<tr>
<th>$C_{L_{max}}$</th>
<th>1.200</th>
<th>1.600</th>
<th>1.750</th>
<th>2.000</th>
<th>2.400</th>
</tr>
</thead>
<tbody>
<tr>
<td>$W/S_{TO}$</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>40.000</td>
<td>0.208</td>
<td>0.156</td>
<td>0.143</td>
<td>0.125</td>
<td>0.104</td>
</tr>
<tr>
<td>60.000</td>
<td>0.313</td>
<td>0.234</td>
<td>0.214</td>
<td>0.188</td>
<td>0.156</td>
</tr>
<tr>
<td>80.000</td>
<td>0.417</td>
<td>0.313</td>
<td>0.286</td>
<td>0.250</td>
<td>0.208</td>
</tr>
<tr>
<td>100.000</td>
<td>0.521</td>
<td>0.391</td>
<td>0.357</td>
<td>0.313</td>
<td>0.260</td>
</tr>
<tr>
<td>120.000</td>
<td>0.625</td>
<td>0.469</td>
<td>0.429</td>
<td>0.375</td>
<td>0.313</td>
</tr>
<tr>
<td>140.000</td>
<td>0.729</td>
<td>0.547</td>
<td>0.500</td>
<td>0.438</td>
<td>0.365</td>
</tr>
<tr>
<td>160.000</td>
<td>0.833</td>
<td>0.625</td>
<td>0.571</td>
<td>0.500</td>
<td>0.417</td>
</tr>
</tbody>
</table>

Table 14. Takeoff Distance Sizing for $S_{TOFL} = 9468$ ft at 5000 ft altitude

<table>
<thead>
<tr>
<th>$C_{L_{max}}$</th>
<th>1.200</th>
<th>1.600</th>
<th>1.750</th>
<th>2.000</th>
<th>2.400</th>
</tr>
</thead>
<tbody>
<tr>
<td>$W/S_{TO}$</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>40.000</td>
<td>0.153</td>
<td>0.115</td>
<td>0.105</td>
<td>0.092</td>
<td>0.077</td>
</tr>
<tr>
<td>60.000</td>
<td>0.230</td>
<td>0.172</td>
<td>0.158</td>
<td>0.138</td>
<td>0.115</td>
</tr>
<tr>
<td>80.000</td>
<td>0.307</td>
<td>0.230</td>
<td>0.210</td>
<td>0.184</td>
<td>0.153</td>
</tr>
<tr>
<td>100.000</td>
<td>0.383</td>
<td>0.287</td>
<td>0.263</td>
<td>0.230</td>
<td>0.192</td>
</tr>
<tr>
<td>120.000</td>
<td>0.460</td>
<td>0.345</td>
<td>0.315</td>
<td>0.276</td>
<td>0.230</td>
</tr>
<tr>
<td>140.000</td>
<td>0.537</td>
<td>0.402</td>
<td>0.368</td>
<td>0.322</td>
<td>0.268</td>
</tr>
<tr>
<td>160.000</td>
<td>0.613</td>
<td>0.460</td>
<td>0.421</td>
<td>0.368</td>
<td>0.307</td>
</tr>
</tbody>
</table>

Table 15. Takeoff Distance Sizing for $S_{TOFL} = 10800$ ft at 8000 ft altitude

<table>
<thead>
<tr>
<th>$C_{L_{max}}$</th>
<th>1.200</th>
<th>1.600</th>
<th>1.750</th>
<th>2.000</th>
<th>2.400</th>
</tr>
</thead>
<tbody>
<tr>
<td>$W/S_{TO}$</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>40.000</td>
<td>0.147</td>
<td>0.110</td>
<td>0.101</td>
<td>0.088</td>
<td>0.074</td>
</tr>
<tr>
<td>60.000</td>
<td>0.221</td>
<td>0.166</td>
<td>0.151</td>
<td>0.133</td>
<td>0.110</td>
</tr>
<tr>
<td>80.000</td>
<td>0.295</td>
<td>0.221</td>
<td>0.202</td>
<td>0.177</td>
<td>0.147</td>
</tr>
<tr>
<td>100.000</td>
<td>0.368</td>
<td>0.276</td>
<td>0.252</td>
<td>0.221</td>
<td>0.184</td>
</tr>
<tr>
<td>120.000</td>
<td>0.442</td>
<td>0.331</td>
<td>0.303</td>
<td>0.265</td>
<td>0.221</td>
</tr>
<tr>
<td>140.000</td>
<td>0.515</td>
<td>0.387</td>
<td>0.353</td>
<td>0.309</td>
<td>0.258</td>
</tr>
<tr>
<td>160.000</td>
<td>0.589</td>
<td>0.442</td>
<td>0.404</td>
<td>0.353</td>
<td>0.295</td>
</tr>
</tbody>
</table>
4.1.3. Landing Distance Sizing

Parameters affecting the landing distance include the landing weight, approach speed, method to decelerate, the pilot’s landing technique, and the quality of the aircraft (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017). Considering that the aircraft is a hybrid, the landing weight will be heavier than conventional transport aircraft. Comparing the conceptual SUGAR aircraft weight ratio of maximum landing weight to takeoff weight, or \( W_L/W_{to} \), provides a strong estimate for hybrid transport weight ratios. Although the fuel depreciates during the stages of the flight, the battery weight does not change, and therefore, the plane will be heavier at landing than an identical fuel-powered aircraft.

The following assumptions will be made for the landing distance sizing: standard conditions, applied brakes to stop the aircraft, and a takeoff weight of 77,602 lbs. Various approach speeds, field lengths, weight ratios, and lift coefficients at landing will determine the possible wing loading values for the HTA.

Using the desired approach speed, the field length can be found.

\[
S_{FL} = 0.3V_A^2
\]  
(28)

From there, the stall speed at landing can be found using:

\[
V_A = 1.3V_{SL}
\]  
(29)

Then, the \((W/S)_L\) results from

\[
(W/S)_L = \frac{V_{SL}^2(0.002378)}{2} \times C_{L_{max}}
\]  
(30)

Thus, for an approach speed of 128 kts, the field length is 4915 ft. The landing stall speed is 98.46 kts. Using a \(C_{L_{max}}\) value of 1.7, the wing loading at landing is 55.8. Then the takeoff wing loading value can be calculated with \(W/S_{to} = \frac{W/S_{L}}{0.96} = 58.2\).
The $W_L/W_{t_0}$ values considered include 0.85, 0.9, 0.93, and 0.96. Together with the varying maximum landing $C_L$ values of 1.7, 2, 2.3, 2.6, and 2.73, and the range of approach speeds from 114 kts to 129 kts, the field length and the wing loading at takeoff were calculated. The summary of the manual calculations is presented in the tables below.

Table 16. W/S to with $W_L/W_{t_0}=0.85$

<table>
<thead>
<tr>
<th>$V_A$ (kts)</th>
<th>$S_{FL}$ (ft)</th>
<th>$C_{L_{maxL}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>1.700</td>
</tr>
<tr>
<td>128.000</td>
<td>4915.200</td>
<td>65.689</td>
</tr>
<tr>
<td>114.900</td>
<td>3960.603</td>
<td>52.931</td>
</tr>
<tr>
<td>115.470</td>
<td>4000.000</td>
<td>53.458</td>
</tr>
<tr>
<td>122.474</td>
<td>4500.000</td>
<td>60.140</td>
</tr>
<tr>
<td>129.099</td>
<td>5000.000</td>
<td>66.822</td>
</tr>
</tbody>
</table>

Table 17. W/S to with $W_L/W_{t_0}=0.9$

<table>
<thead>
<tr>
<th>$V_A$ (kts)</th>
<th>$S_{FL}$ (ft)</th>
<th>$C_{L_{maxL}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>1.700</td>
</tr>
<tr>
<td>128.000</td>
<td>4915.200</td>
<td>62.039</td>
</tr>
<tr>
<td>114.900</td>
<td>3960.603</td>
<td>49.990</td>
</tr>
<tr>
<td>115.470</td>
<td>4000.000</td>
<td>50.488</td>
</tr>
<tr>
<td>122.474</td>
<td>4500.000</td>
<td>56.799</td>
</tr>
<tr>
<td>129.099</td>
<td>5000.000</td>
<td>63.110</td>
</tr>
</tbody>
</table>

Table 18. W/S to with $W_L/W_{t_0}=0.93$

<table>
<thead>
<tr>
<th>$V_A$ (kts)</th>
<th>$S_{FL}$ (ft)</th>
<th>$C_{L_{maxL}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>1.700</td>
</tr>
<tr>
<td>128.000</td>
<td>4915.200</td>
<td>60.038</td>
</tr>
<tr>
<td>114.900</td>
<td>3960.603</td>
<td>48.378</td>
</tr>
<tr>
<td>115.470</td>
<td>4000.000</td>
<td>48.859</td>
</tr>
<tr>
<td>122.474</td>
<td>4500.000</td>
<td>54.966</td>
</tr>
<tr>
<td>129.099</td>
<td>5000.000</td>
<td>61.074</td>
</tr>
</tbody>
</table>
Table 19. W/S<sub>0</sub> with W<sub>L</sub>/W<sub>0</sub>=0.96

\[
C_{L_{\text{max},L}}
\]

<table>
<thead>
<tr>
<th>V&lt;sub&gt;A&lt;/sub&gt; (kts)</th>
<th>S&lt;sub&gt;FL&lt;/sub&gt; (ft)</th>
<th>1.700</th>
<th>2.000</th>
<th>2.300</th>
<th>2.600</th>
<th>2.730</th>
<th>3.000</th>
<th>3.400</th>
</tr>
</thead>
<tbody>
<tr>
<td>128.000</td>
<td>4915.200</td>
<td>58.162</td>
<td>68.426</td>
<td>78.689</td>
<td>88.953</td>
<td>93.401</td>
<td>102.638</td>
<td>116.324</td>
</tr>
<tr>
<td>114.900</td>
<td>3960.603</td>
<td>46.866</td>
<td>55.136</td>
<td>63.407</td>
<td>71.677</td>
<td>75.261</td>
<td>82.705</td>
<td>93.732</td>
</tr>
<tr>
<td>115.470</td>
<td>4000.000</td>
<td>47.332</td>
<td>55.685</td>
<td>64.038</td>
<td>72.390</td>
<td>76.010</td>
<td>83.527</td>
<td>94.664</td>
</tr>
<tr>
<td>122.474</td>
<td>4500.000</td>
<td>53.249</td>
<td>62.646</td>
<td>72.042</td>
<td>81.439</td>
<td>85.511</td>
<td>93.968</td>
<td>106.497</td>
</tr>
<tr>
<td>129.099</td>
<td>5000.000</td>
<td>59.165</td>
<td>69.606</td>
<td>80.047</td>
<td>90.488</td>
<td>95.012</td>
<td>104.409</td>
<td>118.330</td>
</tr>
</tbody>
</table>

4.1.4. Climb Requirement Sizing

In order to size to the climb requirement, the drag polar is necessary to find. The drag coefficient can be summarized as:

\[
C_D = C_{D_0} + \frac{c_d^2}{\pi A e}
\]

(31)

\[
C_{D_0} = f/S
\]

(32)

Since the HTA is modeled after the Boeing 737-100 aircraft, Roskam’s correlation coefficients for parasite area vs. wetted area will be used along with the regression line coefficients for takeoff weight vs. wetted area. These regression coefficients are a= -2.5229, b=1, c=0.0199, and d=0.7531. These coefficients are used in the following equations:

\[
\log_{10}(f) = a + b \times \log_{10}(S_{\text{wet}})
\]

(33)

\[
\log_{10}(S_{\text{wet}}) = c + d \times \log_{10}(W_{\text{to}})
\]

(34)

Using A=8.83, from the Boeing 737-100 model and the e values from Roskam, the drag polars

c

a

n

Table 20. Configuration Parameters and Drag Polars

<table>
<thead>
<tr>
<th>Configuration</th>
<th>(C_{D_0})</th>
<th>A</th>
<th>e</th>
<th>(C_{D_i})</th>
<th>(C_{L_{\text{max}}})</th>
<th>(C_D)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Clean</td>
<td>0.013772</td>
<td>8.83</td>
<td>0.85</td>
<td>0.205265</td>
<td>2.2</td>
<td>0.219037</td>
</tr>
<tr>
<td>Takeoff flaps</td>
<td>0.0334</td>
<td>8.83</td>
<td>0.8</td>
<td>0.137999</td>
<td>1.75</td>
<td>0.171399</td>
</tr>
</tbody>
</table>
Design of a Hybrid Transport Aircraft

These values can be used to calculate the thrust loading at takeoff that meets FAR 25 specifications. The following FAR specifications are for a takeoff climb. For FAR 25.111 (OEI), the configuration consists of gears up, takeoff flaps, ground effects, CGR>0.012 and 1.2\(V_{Sto}\). For FAR 25.121 (OEI), the configuration consists of gears down, takeoff flaps, ground effects, CGR>0, and the speed ranging between \(V_{LOF}\) and 1.2\(V_{Sto}\). For FAR 25.121 (OEI), the configuration consists of gear up, takeoff flaps, no ground effects, and speed at 1.2\(V_{Sto}\). For FAR 121 (OEI), the configuration consists of gears up, flaps up, maximum continuous thrust on remaining engines, CGR>0.012, and speed at 1.25\(V_{S}\) (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017).

In the event of a necessary landing climb from a balked landing, there are two FAR specifications to meet. FAR 25.119 (AEO) requires CGR>0.032, and a configuration of gears down, landing flaps, takeoff thrust on all engines, maximum design landing weight, and 1.3\(V_{SL}\). FAR 25.121 (OEI) requires CGR>0.021, and a configuration of gears down, approach flaps, takeoff thrust on remaining engines, and 1.5\(V_{SA}\). (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017) A summary of these results is presented in the tables below.

Table 21. FAR 25.111 (gears down, TO flaps, with additional T/W calculated for 50°C increase)

<table>
<thead>
<tr>
<th>(C_L) modified</th>
<th>1.215278</th>
</tr>
</thead>
<tbody>
<tr>
<td>(C_D)</td>
<td>0.09995</td>
</tr>
<tr>
<td>L/D</td>
<td>17.50869</td>
</tr>
<tr>
<td>T/W(_{to})</td>
<td>0.138229</td>
</tr>
<tr>
<td>T/W(_{to}) (Modified for +50°C)</td>
<td>0.172786</td>
</tr>
</tbody>
</table>

Table 22. FAR 25.121 (gears down, TO flaps, with additional T/W calculated for 50°C increase)

<table>
<thead>
<tr>
<th>(C_{L,LOF}) ((V=1.1V_{S_{TO}}))</th>
<th>1.446281</th>
</tr>
</thead>
</table>
Design of a Hybrid Transport Aircraft

Table 23. FAR 25.111 (gears down, TO flaps, with additional T/W calculated for 50°C increase)

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_L$ modified</td>
<td>1.215278</td>
</tr>
<tr>
<td>$C_D$</td>
<td>0.09995</td>
</tr>
<tr>
<td>L/D</td>
<td>12.15881</td>
</tr>
<tr>
<td>T/W_to</td>
<td>0.21249</td>
</tr>
<tr>
<td>T/W_to (Modified for +50°C)</td>
<td>0.265612</td>
</tr>
</tbody>
</table>

Table 24. FAR 25.121 (gears up, flaps up, with additional T/W calculated for 50°C increase and maximum thrust at 0.94)

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_L$ modified</td>
<td>1.408</td>
</tr>
<tr>
<td>$C_D$</td>
<td>0.102477</td>
</tr>
<tr>
<td>L/D</td>
<td>13.7397</td>
</tr>
<tr>
<td>T/W_to</td>
<td>0.145437</td>
</tr>
<tr>
<td>T/W_to (Modified for +50°C and 94% thrust)</td>
<td>0.1934</td>
</tr>
</tbody>
</table>

Table 25. FAR 25.119 (AEO) (balked landing, gears down, flaps down)

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_L$ modified</td>
<td>1.615385</td>
</tr>
<tr>
<td>$C_D$</td>
<td>0.218824</td>
</tr>
<tr>
<td>L/D</td>
<td>7.382124</td>
</tr>
<tr>
<td>T/W_to</td>
<td>0.167462</td>
</tr>
<tr>
<td>$W_L$</td>
<td>72946.44</td>
</tr>
<tr>
<td>T/W_to (Modified for +50°C)</td>
<td>0.222689</td>
</tr>
</tbody>
</table>

Table 26. FAR 25.121 (OEI) (balked landing, gears down, flaps down)

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_L$ modified</td>
<td>1.035556</td>
</tr>
<tr>
<td>$C_D$</td>
<td>0.122444</td>
</tr>
</tbody>
</table>
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<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>L/D</td>
<td>8.457407</td>
</tr>
<tr>
<td>T/W_to</td>
<td>0.278479</td>
</tr>
<tr>
<td>W_l</td>
<td>72946.44</td>
</tr>
<tr>
<td>T/W_to (Modified for +50°C)</td>
<td>0.327213</td>
</tr>
</tbody>
</table>

From the data presented in the tables, the rate of climb at altitude and sea level, along with the altitude, absolute ceiling, and other parameters can be found. This information is summarized in Table 27.

Table 27. Sizing to Time to Climb Requirements

<table>
<thead>
<tr>
<th>Sizing to Time to Climb Requirements</th>
<th>( R_C_h )</th>
<th>( R_C_0 )</th>
</tr>
</thead>
<tbody>
<tr>
<td>rate of climb at altitude ( h ) (ft/min)</td>
<td>366.2041</td>
<td>3295.837</td>
</tr>
<tr>
<td>rate of climb at sea level ( R_C ) (ft/min)</td>
<td>( h_{abs} )</td>
<td>45000</td>
</tr>
<tr>
<td>Height (ft)</td>
<td>40000</td>
<td></td>
</tr>
<tr>
<td>absolute ceiling ( h_{abs} ) (ft)</td>
<td>30</td>
<td></td>
</tr>
<tr>
<td>time to climb ( t_{cl} ) (min)</td>
<td>3158.786</td>
<td></td>
</tr>
<tr>
<td>rate of climb ( V ) (ft/min)</td>
<td>957.3788</td>
<td></td>
</tr>
<tr>
<td>T/W (english)</td>
<td>3.377</td>
<td></td>
</tr>
<tr>
<td>L/D</td>
<td>12.8834</td>
<td></td>
</tr>
<tr>
<td>W/S</td>
<td>291.184</td>
<td></td>
</tr>
<tr>
<td>density</td>
<td>0.000738</td>
<td></td>
</tr>
</tbody>
</table>

4.1.5. Maneuvering Requirement Sizing

Maneuvering requirements are generally listed in mission specifications for military aircraft, agrigicultural aircraft, or acrobatic aircraft. Since the HTA’s mission specifications do not include maneuvering capabilities, the aircraft will not be sized to meet these capabilities.

4.1.6. Cruise Speed Requirement Sizing

In order to size to cruise speed requirements, the following equations will be used (Roskam, Airplane Design Part 1: Preliminary Sizing of Airplanes, 2017).

\[
T_{reqd} = C_D \bar{q}S = C_{D_0} \bar{q}S + \frac{c_T \bar{q}S}{\pi A_e}
\]  
\( (36) \)
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\[ W = C_L q S \]  

Combining the two equations by dividing by the weight yields:

\[ \frac{T}{W_{reqd}} = \frac{C_{D_0}}{} \frac{\Delta S}{W} + \frac{C_{L} q S}{W\pi A e} \]  

4.2. AAA Calculation of Performance Constraints

4.2.1. Stall Speed Sizing

<table>
<thead>
<tr>
<th>Input Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>( h_s )</td>
</tr>
<tr>
<td>( V_{C_{max}} )</td>
</tr>
<tr>
<td>( \Delta T_\alpha )</td>
</tr>
<tr>
<td>( V_s )</td>
</tr>
<tr>
<td>( C_{L_{max}} )</td>
</tr>
<tr>
<td>( S_{max} )</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Output Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>( (W/S)<em>{\alpha</em>{max}} )</td>
</tr>
<tr>
<td>( (W/S)<em>{\alpha</em>{min}} )</td>
</tr>
</tbody>
</table>

Figure 31. AAA results of stall speed sizing

4.2.2. Takeoff Distance Sizing

<table>
<thead>
<tr>
<th>Input Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>( h_{TO} )</td>
</tr>
<tr>
<td>( \Delta T_{TO} )</td>
</tr>
<tr>
<td>( C_{L_{max}} )</td>
</tr>
<tr>
<td>( V_{L_{max}} )</td>
</tr>
<tr>
<td>( \Delta C_{L_{max}} )</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Output Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>( (W/S)<em>{\alpha</em>{max}} )</td>
</tr>
</tbody>
</table>

Figure 32. AAA results of takeoff distance sizing

4.2.3. Landing Distance Sizing

<table>
<thead>
<tr>
<th>Input Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>( h_L )</td>
</tr>
<tr>
<td>( W_{L_{max}} )</td>
</tr>
<tr>
<td>( \Delta T_L )</td>
</tr>
<tr>
<td>( C_{L_{max}} )</td>
</tr>
<tr>
<td>( S_L )</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Output Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>( S_{FL} )</td>
</tr>
<tr>
<td>( (W/S)_{L} )</td>
</tr>
</tbody>
</table>


4.2.4. Climb Requirement Sizing

Figure 34. AAA results of climb requirement sizing

4.2.5. Cruise Speed Requirement Sizing

Figure 35. AAA results of cruise speed requirement sizing

4.2.6. Matching Plot
From the performance sizing plot, a design point with a wing loading of 128.34 lb/ft$^2$ and a thrust to weight ratio of 0.31 has been chosen. This yields $C_{L_{max,L}} = 2.73$, $C_{L_{max,T0}} = 2.25$, $S_w = 560.61$ ft$^2$, and a thrust of 21,766 lbs.

5. Fuselage and Cockpit Design

The cockpit and fuselage design will be modeled after the Boeing 737-100 model aircraft. Utilizing Roskam’s *Airplane Design Part 3*, each of these sections will be designed with using typical arrangements that take into account human factors.

5.1. Cockpit Design

The cockpit will be designed for two pilots with standard heights and weight. The cockpit will be initially sized to hold two male crew members, as female crew members are typically smaller. The pilots will be assumed to weigh 175 lbs, based on body component weights and the fact that they will not be wearing a helmet. For dimensions, the body width consists of the shoulder width, which will be estimated at 21 in. across, the width at the elbows, estimated at 22 in. across, and the hip width at 18 in. across. The weights and dimensions include flight clothing. The remaining dimension assumptions are illustrated in Figure 37, below.
The cockpit design must also take into account the visibility during takeoff, cruise, and landing. Figure 37 also illustrates the angles for line of sight. Roskam defines cockpit visibility as “the angular area obtained by intersecting the airplane cockpit with radial vectors emanating from the eyes of the pilot” (Roskam, 2011). The point C, in Figure 38, lies on the eyes of the pilot. Figure 39 demonstrates the minimum visibility pattern for transport aircraft, per FAR 25 specifications.
The cockpit layouts in section 5.1.1. meet the visibility requirements and height, width, and weight requirements for the pilots. They are from the Boeing 737-100 model since the exchange of the hybrid engine, and battery nacelles will not change the fuselage nor cockpit design.
5.1.1. Cockpit Layout Design

Figure 40. Side and top-down view of cockpit. Dimensions in meters (Boeing Commercial Airplanes, 2013)

Figure 41. Front and side 3-D view of cockpit. Dimensions in meters (Boeing Commercial Airplanes, 2013)
5.2. Fuselage Design

5.2.1. Fuselage Layout Design

*NOTES:*
- SIX-ABREAST SEATING
- 103 PASSENGERS AT 34−IN (0.86 M) PITCH AS SHOWN
  OR 118 PASSENGERS AT 30−IN (0.76 M) PITCH
- GALLEY AFT

*Figure 42.* Rear view of cockpit. (*Aircraft Technical Data and Specifications: Boeing 737-100/200, 2017*)

*Figure 43.* Top down view of the fuselage (Boeing Commercial Airplanes, 2013)
Figure 44. Front view of the fuselage (Boeing Commercial Airplanes, 2013)

Figure 45. Front 3-D view of the 6 abreast passenger seating in the fuselage (Creative Aviation Photography, 2017)
Figure 46. Rear 3-D view of the 6 abreast passenger seating in the fuselage (Creative Aviation Photography, 2017)
Figure 47. Side view of fuselage with cargo compartment visible (Boeing Commercial Airplanes, 2013)
5.3. Recommendations and Conclusions

5.3.1. Recommendations

In general, the HTA is a hybrid version of the Boeing 737-100. Considering the 96 passenger payload, the 6 abreast seating allows for a smaller fuselage and lower takeoff weight. The standard cockpit design will work with two pilots, whether male or female. The fuselage design allows for flight attendant seating.

5.3.2. Conclusions

The current model of the 737-100 fuselage and cockpit meets FAR 25 requirements, visibility requirements, and provides the opportunity to exchange the engine and add batteries. To keep the takeoff weight a minimum, the minimum number of passengers will be considered, and the plane can be expanded length-wise in a future design.
6. Wing, High-Lift System and Lateral Control Design

6.1. Wing Planform Design

Using the known gross wing area, $S$, and aspect ratio, $A$, the remaining planform design characteristics of the wing will be chosen. These characteristics include the sweep angle, $\Lambda$, thickness ratio, $t/c$, airfoil type, taper ratio, $\lambda$, incidence angle $i_w$, twist angle, $\varepsilon_w$, dihedral angle, $\Gamma_w$, and the lateral control surface size and layout. Some of these will be calculated, and others will be chosen based on reference data. The HTA will feature conventional, cantilever low wing.

6.1.1. Sweep Angle-Thickness Ratio Combination

The 737-100 has a wing quarter chord sweep angle of 25º and is, therefore, aft swept. The HTA will feature the same 25 º sweep angle. This angle also matches well with other jet transport sweep angles, visible in the Figure 49 below.

![Figure 49. Jet Transports Wing Geometric Data (Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System, 2011)](image-url)
The next values to select for the wing parameters are the taper ratio and the dihedral angle. The taper ratio consists of the ratio of the tip chord to root chord. The dihedral angle is the upward angle from the horizontal to the tailplane of the fixed-wing aircraft. These values will be modeled after the Boeing 737-100. \( \lambda = 0.339 \) and \( \Gamma_w = 6. \)

Since the airplane will be flying at high subsonic speeds, a trade study will be performed varying the sweep angle and the thickness ratio. This will facilitate the choice of the sweep angle and thickness ratio that allows for the minimum weight of the wing. The following equation:

\[
\frac{M_{cc} \cos^2 \Lambda}{\sqrt{1 - M_{cc}^2 \cos^2 \Lambda}} \left( \frac{2.64(\frac{\Gamma}{\Lambda})}{\cos \Lambda} \right) + \left( \frac{\gamma+1}{2} \right) \left( \frac{2.64(\frac{\Gamma}{\Lambda})(0.34c_j)}{\cos^3 \Lambda} \right) + \frac{M_{cc} \cos^2 \Lambda}{1 - M_{cc}^2 \cos^2 \Lambda} \left[ \left( \frac{\gamma+1}{2} \right) \left( \frac{1.32(\frac{\Gamma}{\Lambda})}{\cos \Lambda} \right)^2 + \frac{M_{cc} \cos^2 \Lambda}{1 - M_{cc}^2 \cos^2 \Lambda} \left[ \left( \frac{\gamma+1}{2} \right) \left( \frac{0.68c_j}{\cos \Lambda} \right) + \left( \frac{\gamma+1}{2} \right) \left( \frac{0.34c_j}{\cos \Lambda} \right) \right] - 1 = 0 \right.
\]

(39)

can be used to solve for the thickness ratio for varying sweep angles. The table and plot below demonstrate these results.

**Table 28. Thickness Ratio and Wing Weight vs. sweep angle**

<table>
<thead>
<tr>
<th>( \Lambda ) (deg)</th>
<th>( t/c )</th>
<th>( W_w ) (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0.079</td>
<td>( \text{---------} )</td>
</tr>
<tr>
<td>5</td>
<td>0.081</td>
<td>436448.0332</td>
</tr>
<tr>
<td>10</td>
<td>0.086</td>
<td>81961.74273</td>
</tr>
<tr>
<td>15</td>
<td>0.095</td>
<td>44021.92291</td>
</tr>
<tr>
<td>20</td>
<td>0.108</td>
<td>29471.24401</td>
</tr>
<tr>
<td>25</td>
<td>0.124</td>
<td>21783.48926</td>
</tr>
<tr>
<td>30</td>
<td>0.145</td>
<td>17064.19453</td>
</tr>
<tr>
<td>35</td>
<td>0.169</td>
<td>13911.69355</td>
</tr>
<tr>
<td>40</td>
<td>0.197</td>
<td>11691.97308</td>
</tr>
<tr>
<td>45</td>
<td>0.227</td>
<td>10076.63814</td>
</tr>
<tr>
<td>50</td>
<td>0.259</td>
<td>8877.443908</td>
</tr>
<tr>
<td>55</td>
<td>0.291</td>
<td>7981.378457</td>
</tr>
<tr>
<td>60</td>
<td>0.319</td>
<td>7320.729094</td>
</tr>
<tr>
<td>65</td>
<td>0.338</td>
<td>6862.08962</td>
</tr>
<tr>
<td>70</td>
<td>0.337</td>
<td>6614.330935</td>
</tr>
<tr>
<td>75</td>
<td>0.295</td>
<td>6700.319668</td>
</tr>
<tr>
<td>80</td>
<td>0.142</td>
<td>8184.063457</td>
</tr>
<tr>
<td>85</td>
<td>-0.454</td>
<td>( \text{---------} )</td>
</tr>
</tbody>
</table>

**Figure 50.** Plots of \( t/c \) and \( W_w \) vs. \( \Lambda \)
Using the wing weight as the figure of merit, the minimum wing weight occurs at a sweep angle of 70°. This is not a reasonable sweep angle, and therefore, the wing weight may not be an appropriate figure of merit.

Besides using the weight of the wing as a figure of merit, there are other considerations for selecting the thickness ratio. The thickness ratio should be greater than 0.1 to accommodate the wing structure, fuel, batteries, and main landing gear when retracted. Additionally, the thickness ratio should not exceed 0.2 in order to prevent the drag profile from being too high. Additionally, large sweep angles result in lower performance of the wing during low speed. As a result, high lift systems may be necessary to improve this performance. Using these values for the thickness ratio, a sweep angle between 20 and 40° would be appropriate. Thus, the 25° angle will be selected and this results in a thickness ratio of 0.124.

6.2. Airfoil Selection

Selecting the airfoil involves considering the aircraft mission, along with properties of the drag coefficient at the design lift coefficient, section critical Mach number, and section pitching moment. The Boeing 737-100 used the Wing root- BAC 449/450/451 airfoil with Wing tip- BAC 442 airfoil. This will be selected as well for the HTA. The schematic, below, illustrates the various airfoils for the root airfoil, two midspan airfoils, and outboard airfoil.

![Boeing 737 Airfoil Sections](image)

*Figure 51. Boeing 737 Airfoil selections schematic (Aircraft Technical Data and Specifications: Boeing 737-100/200, 2017)*
Based on Roskam’s wing geometric data for transport aircraft, the incidence angle will be 1° (Aircraft Technical Data and Specifications: Boeing 737-100/200, 2017). Using the Boeing 737-100 model, the dihedral angle used will be 6°. Most aircraft have negative wash-out twist angles. Based on Roskam’s discussion of designing wings for transport aircraft, a twist angle of -2° will be assumed.

6.3. Design of the High-Lift Devices

High lift devices are useful to help the aircraft meet its requirements for $C_{z0}$ and $C_{z0}$ and $C_{z0}$. High lift devices include leading-edge devices and trailing-edge devices. Leading edge devices include VC Krueger flap, two-position slat, three-position slat, fixed slot, variable camber (VC) leading edge, hinged leading edge (or droop nose), folding bull-nose Krueger flap (Rudolph, 1996). Trailing edge devices include split flap, plain flap, simple slotted flap, single-slotted Fowler flap, fized vain/main slotted flap, articulating vane/main double slopped flap, main/aft double slotted flap, triple-slotted flap (Rudolph, 1996).

The following sizing will determine which high lift devices would provide the HTA with enough lift to meet the takeoff and landing mission requirements. The HTA has $C_{l_{max}}$ = 2.73 and $C_{l_{maxTO}}$ = 2.25. These values were determined from the performance sizing in discussed in Chapter 4. According to Roskam, sizing the high lift devices involves determining the maximum lift coefficient capability of the wing, estimating $\Delta C_{l_{maxTO}}$ and $\Delta C_{l_{maxL}}$, accounting for the sweep of the wing, and then sizing the device using the desired $\Delta C_{l}$. Using the planform design characteristics of $\Lambda$=25°, $t/c=0.124$, $\lambda=0.339$, $i_w=2°$, $\epsilon_w=\Gamma=6°$, $S=1098$ ft², $b=93$ ft, $c_r=24.02$ ft, and $c_t=5.2$ ft, the high lift devices will be sized.

The $C_{l_{max}}$ can be obtained from the $C_{l_{maxW}}$. To calculate the maximum lift coefficient of the wing, the following equation will be used:
Thus, using \( k_\lambda = 0.957 \) and obtaining the root and tip maximum lift coefficients for the airfoil, \( C_{l_{\text{max}}w} \) can be calculated, with \( C_{l_{\text{max}}w} = \frac{0.957 (1.9 + 1.6)}{2} = 1.67 \).

\[
C_{l_{\text{max}}w} = \frac{k_\lambda (C_{l_{\text{maxr}}} + C_{l_{\text{maxr}}})}{2} \tag{40}
\]

Figure 52. \( C_l \) vs. angle of attack for maximum lift coefficient at root and midspan

\( C_{l_{\text{max}}w} = 1.1 \), and taking into account the quarter chord sweep angle, \( C_{l_{\text{max}}w} = 1.8 \).

Since the original 737-100 is a short-coupled airplane, the HTA will be assumed to be a short-coupled airplane. Therefore,

\[
C_{l_{\text{max}}w} = 1.1 C_{l_{\text{max}}} \tag{41}
\]

and \( C_{l_{\text{max}}} = 1.6 \).

Using \( C_{l_{\text{max}}} = 1.6 \), the \( \Delta C_{l_{\text{max}}w} \) and \( \Delta C_{l_{\text{max}}L} \) can be calculated using the following two equations.

\[
\Delta C_{l_{\text{max}}w} = 1.05 (C_{l_{\text{maxr}}} - C_{l_{\text{maxr}}}) \tag{42}
\]

\[
\Delta C_{l_{\text{max}}L} = 1.05 (C_{l_{\text{maxL}}} - C_{l_{\text{maxL}}}) \tag{43}
\]

Therefore, \( \Delta C_{l_{\text{max}}w} = 1.05 (2.25 - 1.1) = 1.2 \) and \( \Delta C_{l_{\text{max}}L} = 1.05 (2.73 - 1.1) = 1.7 \). The factor of 1.05 takes into account the trip penalties incurred when using flaps. To calculate the
ΔC_{l_{\text{max}}} values, with the sweep correction factor taken into consideration, the following table
illustrates the various ΔC_{l_{\text{max}}} with three arbitrary flapped wing area to wing area ratios.

Table 29. ΔC_{l_{\text{max,TO}}} and ΔC_{l_{\text{max,L}}} using various S_{wf}/S

<table>
<thead>
<tr>
<th>S_{wf}/S</th>
<th>ΔC_{l_{\text{max,TO}}}</th>
<th>ΔC_{l_{\text{max,L}}}</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.60</td>
<td>0.83</td>
<td>1.18</td>
</tr>
<tr>
<td>0.80</td>
<td>1.11</td>
<td>1.57</td>
</tr>
<tr>
<td>0.90</td>
<td>1.25</td>
<td>1.77</td>
</tr>
</tbody>
</table>

In order to achieve the required flap lift increments, Fowler flaps will be used. The Fowler flaps have the following geometry: \( \frac{c_f}{c} = 0.3, \delta_{f0} = 35^\circ, \delta_{fL} = 40^\circ \). (Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System, 2011). The required incremental section lift coefficient, ΔC₁ will be calculated. This is the value that the trailing edge flaps must generate. The ΔC_{l_{\text{max}}} values have been estimated at takeoff and landing based on the S_{wf}/S ratio. ΔC₁ is related to ΔC_{l_{\text{max}}} through the following equation

\[
\Delta C_1 = \left( \frac{1}{K} \right) \Delta C_{l_{\text{max}}}
\]

(44)

K=0.94 is determined using , below, with \( c_f/c = 0.3 \) Thus the needed ΔC₁ values can be summarized in Table 30.

![Figure 53. Effect of Flap Chord Ratio and Flap Type on K](image)
Table 30. Calculated $\Delta c_l$ for takeoff and landing

<table>
<thead>
<tr>
<th>$S_{wf}/S$</th>
<th>$\Delta c_{l_{to}}$</th>
<th>$\Delta c_{l_{li}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.60</td>
<td>0.88</td>
<td>1.26</td>
</tr>
<tr>
<td>0.80</td>
<td>1.18</td>
<td>1.67</td>
</tr>
<tr>
<td>0.90</td>
<td>1.33</td>
<td>1.88</td>
</tr>
</tbody>
</table>

For Fowler flaps, the following equations hold:

$$\Delta c_l = c_l a \delta_f$$  \hspace{1cm} (45)

$$c_{l_{af}} = c_{l_a} c'/c$$  \hspace{1cm} (46)

$$\frac{c'}{c} = 1 + 2 \left( \frac{2 f_k}{c} \right) \tan \left( \frac{\delta L}{2} \right)$$  \hspace{1cm} (47)

Therefore, $\Delta c_l = 2\pi (0.53) \left( \frac{35}{57.29} \right) = 2.03$. Finally, using

$$\frac{S_{wf}}{S} = (\eta_0 - \eta_i)[2 - (1 - \lambda)(\eta_i + \eta_0)]/(1 + \lambda)$$  \hspace{1cm} (48)

the following is a summary of the flap geometry: $\frac{S_{wf}}{S} = 0.83$, $\frac{c_f}{c} = 0.3$, $\delta_{f_{to}} = 35^\circ$, $\delta_{f_{li}} = 40^\circ$.

The $\Delta c_l$ supplied by the Fowler flaps satisfies the required $\Delta c_l$ for both takeoff and landing. Thus, the Fowler flaps will be sufficient in providing additional lift.

6.3.1. AAA Calculation of Lift Coefficient

Using AAA, the clean maximum lift coefficient was found to be 1.541. Although this value is lower than the original manual calculations, the increase in lift provided by the Fowler Flaps is still substantial to provide the aircraft with lift.

Figure 54. AAA Estimate of clean Lift Coefficient of the Wing
6.4. Design of the Lateral Control Surfaces

Roskam provides reference data for the lateral control surfaces. The data for jet transports is summarized in the table below. The aircraft will have a rudder area of 56.2 ft\(^2\), a fin height of 20.2 ft, and a fin area of 224 ft\(^2\). The HTA will also have ailerons and spoilers with the ailerons composing 2.4% of the wing area.

In summary, the HTA will have the following wing parameters: b=93 ft, c=24 ft, c=5.2 ft, mac=12.5 ft, mgc=12.5 ft, \(\Lambda_{LE}=27.8^\circ\), \(\Lambda_{TE}=16.625^\circ\), and \((40.7, 43.8)\)= coordinates of the aerodynamic center measured in feet from the center of gravity.

The following figures will illustrate the wing planform and show these parameters.

*Table 31. Jet Transport Lateral Control Surface Information*
### Table 3.2. Additional Jet Transport Lateral Control Surface Information

<table>
<thead>
<tr>
<th>Type</th>
<th>Outb’d All.</th>
<th>Outb’d Spoiler Span</th>
<th>Inb’d All.</th>
<th>Inb’d Spoiler Span</th>
<th>Inb’d All.</th>
<th>Inb’d Spoiler Span</th>
<th>Outb’d All.</th>
<th>Outb’d Spoiler Span</th>
<th>Inb’d All.</th>
<th>Inb’d Spoiler Span</th>
<th>Outb’d All.</th>
<th>Outb’d Spoiler Span</th>
<th>Inb’d All.</th>
<th>Inb’d Spoiler Span</th>
</tr>
</thead>
<tbody>
<tr>
<td>737-300</td>
<td>.72/.91</td>
<td>.23/.30</td>
<td>.38/.44</td>
<td>0.14</td>
<td>.64/.70</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>747-200B</td>
<td>.70/.93</td>
<td>.11/.17</td>
<td>.46/.67</td>
<td>.12/.16</td>
<td>.91</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>747-SF</td>
<td>.70/.93</td>
<td>.11/.17</td>
<td>.46/.67</td>
<td>.12/.16</td>
<td>.91</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>757-200</td>
<td>.76/.97</td>
<td>.22/.36</td>
<td>.41/.74</td>
<td>.12/.13</td>
<td>.73/.69</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>McDonnell-Douglas</td>
<td>DC-9 580</td>
<td>.64/.45</td>
<td>.31/.56</td>
<td>.35/.60</td>
<td>.10/.08</td>
<td>.69/.45</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>.65/.47</td>
<td>.12/.16</td>
<td>none</td>
</tr>
<tr>
<td>DC-9-50</td>
<td>.78/.93</td>
<td>.30/.35</td>
<td>.35/.60</td>
<td>.10/.08</td>
<td>.69/.45</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>DC-10-30</td>
<td>.75/.93</td>
<td>.29/.27</td>
<td>.17/.50</td>
<td>.05/.06</td>
<td>.78/.74</td>
<td>.43/.72</td>
<td>.11/.16</td>
<td>.75/.70</td>
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<td></td>
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<td></td>
<td></td>
</tr>
<tr>
<td>Airbus</td>
<td>A330-300</td>
<td>.83/.59</td>
<td>.32/.50</td>
<td>.16/.22</td>
<td>.71/.72</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>A310</td>
<td>none</td>
<td>none</td>
<td>.62/.33</td>
<td>.16/.22</td>
<td>.69/.66</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fokker F-28</td>
<td>.66/.91</td>
<td>.23/.28</td>
<td>no lateral control spoilers</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
<td>none</td>
</tr>
</tbody>
</table>

- Rombac/British Aerospace
- 7-11 495 | .72/.92 | 0.26 | .57/.68 | .06/.21 | .64/.69 | none | none | none | none | none | none | none | none | none |
- British Aerospace
- 146-200 | .76/.94 | .53/.31 | .14/.70 | .22/.27 | .76/.68 | none | none | none | none | none | none | none | none | none |
- Tu-154 | .76/.98 | .34/.27 | .43/.70 | .14/.20 | .62/.60 | none | none | none | none | none | none | none | none | none |
6.5. Drawings

![Wing Planform: flap and lateral control layout](image)

Figure 55. Wing Planform: flap and lateral control layout

6.6. Discussion

The HTA planform and lateral control layout illustrate the wing parameters. The Fowler flaps will be full span, and the spoilers will be used for lateral control. The spoilers will run from \(0.4\left(\frac{b}{2}\right)\) to \(0.66\left(\frac{b}{2}\right)\) with 0.14\(c\) to 0.18\(c\). The outboard ailerons will run from \(0.74\left(\frac{b}{2}\right)\) to \(0.94\left(\frac{b}{2}\right)\) with 0.20\(c\) to 0.28\(c\). Additionally, the drawings demonstrate the span of 93 feet, and the root and tip chord.
6.7. Recommendations and Conclusions

6.7.1. Recommendations

Based on the geometry of the wing and the flight parameters chosen, Fowler flaps are the recommended high lift device that will provide substantial lift. Additionally, the fuel can be stored in the wing without the risk of fire due to lightning strikes since the fuel in the HTA is less than the fuel in the 737-100. The batteries will additionally be suspended below the wing in nacelles, and this may change the aerodynamics surrounding the flaps and the nacelles.

6.7.2. Conclusions

The data in both tables from Roskam, along with pre-existing Boeing 737-100 specifications, provides benchmarks with which to compare the wing parameters. Although wing weight is usually a figure of merit for the thickness ratio, it proved to be a poor basis for the thickness ratio. Using a thickness ratio greater than 0.1 and less than 0.2 proved to provide optimal sweep angles of 25° at the quarter chord, along with a reasonable thickness ratio of 0.124. Additionally, the 25° sweep angle, which is the same as the original 737-100, yielded \( \Lambda_{LE} = 27.8^\circ \), \( \Lambda_{TE} = 16.625^\circ \), from linear interpolation.

Through the analysis of the Boeing airfoil data, the \( \Delta C_{z_{max}} \) was calculated, and the Fowler flaps were found to provide substantial lift, without any leading edge devices. The HTA has requirements for \( C_{z_{max_{to}}} = 2.25 \) and \( C_{z_{max_{L}}} = 2.73 \). The calculations demonstrated that an additional \( \Delta C_{z_{max}} \) of 1.25 is needed for takeoff and \( \Delta C_{z_{max}} \) of 1.77 is needed for landing. Since the Fowler flaps provide an additional \( \Delta C_{z_{max}} = 2.03 \), they are substantial in providing lift.

Other high lift devices could be considered for the HTA. Given the geometry of the wing and the parameters chosen, the Fowler flaps provide adequate lift, and the wing geometry provides adequate storage for the fuel.
7. Design of the Empennage and the Longitudinal and Directional Controls

7.1. Overall Empennage Design

The empennage design, which was initially introduced in the configuration design chapter is composed of the tail and stabilizing features organized in a tail aft arrangement. The HTA will feature a single vertical tail mounted to the fuselage along with a horizontal stabilizer.

For conventional configurations, the center of mass is generally near wing’s aerodynamic center. This is the point at which the wing’s pitching moment coefficient doesn’t vary with changes in angle of attack. The empennage moment arms, $x_h, x_v, x_c$, will be located so as to create a maximum moment arm relative to the center of gravity (Roskam, Airplane Design Part III: Layout Design of Cockpit, Fuselage, Wing, and Empennage: Cutaways and Inboard Profiles, 2011). Besides creating a large moment arm, the placement of these locations minimizes the area of the empennage. This helps minimize the weight of the aircraft and the drag during flight. For the HTA, $x_h = 43.8\text{ ft and } x_v = 40.7\text{ ft}$. The in the table below summarizes the volume coefficients and control surface data.

Table 33. Summary of Comparable Volume Coefficients and Control Surface Data

<table>
<thead>
<tr>
<th>Airplane Type</th>
<th>$\bar{V}_h$</th>
<th>$S_e/S_h$</th>
<th>$\bar{V}_v$</th>
<th>$S_r/S^v$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boeing 737-200</td>
<td>1.28</td>
<td>0.27</td>
<td>0.100</td>
<td>0.24</td>
</tr>
<tr>
<td>Boeing 737-300</td>
<td>1.35</td>
<td>0.24</td>
<td>0.100</td>
<td>0.31</td>
</tr>
<tr>
<td>McDD DC-9-S80</td>
<td>0.96</td>
<td>0.34</td>
<td>0.062</td>
<td>0.39</td>
</tr>
<tr>
<td>McDD DC-9-50</td>
<td>1.32</td>
<td>0.38</td>
<td>0.079</td>
<td>0.41</td>
</tr>
<tr>
<td>Fokker F-28-4000</td>
<td>1.07</td>
<td>0.20</td>
<td>0.085</td>
<td>0.16</td>
</tr>
<tr>
<td>Rombac/Bae 1-11-495</td>
<td>0.86</td>
<td>0.27</td>
<td>0.038</td>
<td>0.28</td>
</tr>
<tr>
<td>Averages:</td>
<td>1.14</td>
<td>0.28</td>
<td>0.077</td>
<td>0.30</td>
</tr>
</tbody>
</table>

$$S_h = \bar{V}_h S\bar{c}/x_h$$  \hfill (49)
Together, these will compose the horizontal stabilizer and the vertical stabilizer.

7.2. Design of the Horizontal Stabilizer

The horizontal stabilizer area is 401 ft$^2$, and was calculated using equation 1:

\[
S_h = \frac{\bar{V}Sb}{x_h} = \frac{(128)(1098)(12.5)}{43.8} = 401 \text{ ft}^2.
\]

Additional parameters of the horizontal stabilizer include the aspect ratio, quarter chord sweep angle, taper ratio, thickness ratio, airfoil, dihedral angle, and incidence angle. The 737-100 model will be used as a reference for these parameters. Therefore, the horizontal stabilizer aspect ratio will be 4.15. The thickness ratio will be 0.12. The horizontal stabilizer will have a 7º dihedral angle, a taper ratio of 0.26, a quarter chord sweep angle of 30º and a variable incidence angle (Aircraft Technical Data and Specifications: Boeing 737-100/200, 2017).

The horizontal stabilizer will span 36 feet and will have a tailplan area of 312 ft$^2$ and an elevator area of 70.5 ft$^2$. The elevator is 0.3 of the chord at the root and 0.32 of the chord at the tip. The airfoil used is the BAC airfoil by Boeing (Aircraft Technical Data and Specifications: Boeing 737-100/200, 2017).

7.3. Design of the Vertical Stabilizer

The vertical stabilizer, or fin, provides directional stability, and will have an area of

\[
S_v = \frac{\bar{V}Sb}{x_v} = \frac{(0.1)(1098)(93.011)}{40.7} = 250.9 \text{ ft}^2.
\]

Additionally, using the 737-100 for reference, the design parameters of the vertical stabilizer will be chosen. These include the aspect ratio of 1.64, the quarter chord sweep angle of 35º, a taper ratio of 0.288, a thickness ratio of 0.15, and a 0º incidence angle. Additionally, the fin height of 20 feet and a rudder area of 56 ft$^2$. The airfoil used is the BAC airfoil by Boeing (Aircraft Technical Data and Specifications: Boeing 737-100/200, 2017).
7.4. Empennage Configuration Drawings

The figure below consists of the drawings of the empennage arrangement for the HTA. Included in this drawing are the elevator and rudder outlines. All dimensions are in feet and are illustrated below.

Figure 56. Empennage Configuration (units in feet)
7.5. Empennage Design Evaluation

AAA will be used to analyze the performance of the empennage horizontal and vertical stabilizers. Using the inputs of AP, $\lambda$, S, $\Lambda_{c/4}$, and the x, y, and z offset values, the output parameters were calculated. This provided the root and tip dimensions, span, mean geometric chord, leading edge and trailing edge sweep angle, and the positions for the mean geometric chord for both the vertical and horizontal tail. The outputs and inputs are presented below.

<table>
<thead>
<tr>
<th>Input Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>$AR_{h}$</td>
</tr>
<tr>
<td>$h_{h}$</td>
</tr>
<tr>
<td>$\Lambda_{40_{h}}$</td>
</tr>
<tr>
<td>$\Theta_{h}$</td>
</tr>
<tr>
<td>$\Lambda_{c/4}$</td>
</tr>
<tr>
<td>$Y_{off_{h}}$</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Output Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{h}$</td>
</tr>
<tr>
<td>$b_{h}$</td>
</tr>
<tr>
<td>$\Lambda_{rep_{h}}$</td>
</tr>
<tr>
<td>$\Lambda_{s_{h}}$</td>
</tr>
<tr>
<td>$C_{v}$</td>
</tr>
<tr>
<td>$b_{v}$</td>
</tr>
<tr>
<td>$\Lambda_{rep_{v}}$</td>
</tr>
<tr>
<td>$\Lambda_{s_{v}}$</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Straight Tapered Horizontal Tail Geometry: Output Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>Panel</td>
</tr>
<tr>
<td>1</td>
</tr>
</tbody>
</table>

Figure 57. Horizontal Tail Geometry Calculations by AAA

<table>
<thead>
<tr>
<th>Input Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>$AR_{v}$</td>
</tr>
<tr>
<td>$h_{v}$</td>
</tr>
<tr>
<td>$\Lambda_{40_{v}}$</td>
</tr>
<tr>
<td>$\Theta_{v}$</td>
</tr>
<tr>
<td>$\Lambda_{c/4}$</td>
</tr>
<tr>
<td>$\Lambda_{rep_{v}}$</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Output Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{v}$</td>
</tr>
<tr>
<td>$b_{v}$</td>
</tr>
<tr>
<td>$\Lambda_{rep_{v}}$</td>
</tr>
<tr>
<td>$\Lambda_{s_{v}}$</td>
</tr>
<tr>
<td>$C_{v}$</td>
</tr>
<tr>
<td>$b_{v}$</td>
</tr>
<tr>
<td>$\Lambda_{rep_{v}}$</td>
</tr>
<tr>
<td>$\Lambda_{s_{v}}$</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Straight Tapered Vertical Tail Geometry: Output Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>Panel</td>
</tr>
<tr>
<td>1</td>
</tr>
</tbody>
</table>

Figure 58. Vertical Tail Geometry Calculations by AAA
7.6. Discussion

Since the use of batteries during various stage of flight will not change the empennage, the 737-100 model was used as a reference for the HTA. This data, along with similar data provided by Roskam, provided the foundation for initial selections for the parameters. Additionally, the parameters chosen are consistent with the control surface ratios provided by Roskam. Thus, $S_e/S_h = 0.27$ for a 737-200 and $S_e/S_h = 0.24$ for a 737-300. Using the 737-100 parameters, $S_e/S_h = \frac{70.5}{401} = 0.18$. This value is lower, but comparable. For the rudder area to vertical stabilizer area, $\frac{S_r}{S_v} = 0.31$ and 0.39 for 737-200 and 737-300, respectively. For the HTA, $\frac{S_r}{S_v} = \frac{56}{250.9} = 0.22$. Again, although this is lower, it is comparable.

AAA additionally calculated very similar values to the original 737-100 aircraft. The tail geometry is consistent with previous aircraft, and will provide adequate stability for the HTA.

7.7. Recommendations and Conclusions

7.7.1. Recommendations

The HTA empennage will be modeled after the 737-100 airplane with slight changes. The moment arms have been estimated at $x_h = 43.8$ ft and $x_v = 40.7$ ft. These locations will be iterated to determine the aerodynamic center that provides the best stability and control.

7.7.2. Conclusions

The tail aft empennage arrangement provides a suitable conventional configuration design for the HTA. With the hybrid plane being modeled after the 737-100, the parameters have been chosen based on the fuel powered model. The distance from the cg to the ac will be established further when exploring the stability and control of the landing gear and may be adjusted, if necessary, after an iterative process. The figures below illustrate how the empennage fits in the aircraft.
**Figure 59.** Vertical tail configuration in HTA

**Figure 60.** Horizontal tail configuration in HTA
8. Design of the Landing Gear and Weight and Balance Analysis

8.1. Introduction

The landing gear arrangement, previously introduced in the configuration chapter, will consist of retractable conventional tricycle landing gear. The main landing gear will retract into the fuselage, and the nose landing gear will retract into the nose. In order to size the landing gear, the center of gravity will need to be determined. From there, a weight and balance iteration will be performed to determine whether the cg of the proposed design is in the correct location. The cg excursion diagram will be used to illustrate the possible locations of the cg.

8.2. Estimation of the Center of Gravity

The following data consists of the component weight and coordinate data for the Class I component weight estimation Assuming a takeoff weight of 83,000 lbs and the weight fractions from Roskam’s Part V. Component Weight Estimation, the weights and positions are summarized in the table below (Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System, 2011).

The zero reference point will be located a distance well to the left and below the airplane. This allows the airplane to change size without changing the reference point. The x-axis will refer to x-coordinates called the fuselage stations (F.S.) and the y-axis coordinates will be referred to as the wing buttock lines (B.L.). For the HTA, the first F.S. point will be at 151 inches, and the first B. L. point will be at 78 inches.

The comcomponent weight and coordinate data for the HTA are summarized in Table 34, below. Note that the aircraft is symmetric, so the y coordinates will all be 0 inches.
### Table 34. HTA Component Weight and Coordinate Data

<table>
<thead>
<tr>
<th>No.</th>
<th>Type of Component</th>
<th>$W_i$ (lbs)</th>
<th>$x_i$ (in.)</th>
<th>$W_i x_i$ (in lbs)</th>
<th>$y_i$ (in.)</th>
<th>$W_i y_i$ (in lbs)</th>
<th>$z_i$ (in.)</th>
<th>$W_i z_i$ (in lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Fuselage Group</td>
<td>8,549</td>
<td>641</td>
<td>5,479,909</td>
<td>0</td>
<td>0</td>
<td>248</td>
<td>2,120,152</td>
</tr>
<tr>
<td>2</td>
<td>Wing Group</td>
<td>7,636</td>
<td>680</td>
<td>5,192,480</td>
<td>0</td>
<td>0</td>
<td>350</td>
<td>2,672,600</td>
</tr>
<tr>
<td>3</td>
<td>Empennage Group</td>
<td>1,992</td>
<td>1190</td>
<td>2,374,480</td>
<td>0</td>
<td>0</td>
<td>300</td>
<td>597,600</td>
</tr>
<tr>
<td>4</td>
<td>Engine Group</td>
<td>20,950</td>
<td>590</td>
<td>12,360,500</td>
<td>0</td>
<td>0</td>
<td>140</td>
<td>2,933,000</td>
</tr>
<tr>
<td>5a</td>
<td>Landing Gear Group: Nose Gear</td>
<td>500</td>
<td>390</td>
<td>195,000</td>
<td>0</td>
<td>0</td>
<td>130</td>
<td>74,750</td>
</tr>
<tr>
<td>5b</td>
<td>Landing Gear Group: Main Gear</td>
<td>2654</td>
<td>710</td>
<td>1,884,340</td>
<td>0</td>
<td>0</td>
<td>130</td>
<td>345,020</td>
</tr>
<tr>
<td>6</td>
<td>Fixed Equipment Group</td>
<td>10,707</td>
<td>680</td>
<td>7,280,760</td>
<td>0</td>
<td>0</td>
<td>200</td>
<td>2,141,400</td>
</tr>
<tr>
<td>7</td>
<td>Trapped Fuel and Oil</td>
<td>142.8</td>
<td>690</td>
<td>98,532</td>
<td>0</td>
<td>0</td>
<td>215</td>
<td>30,702</td>
</tr>
<tr>
<td>8</td>
<td>Crew</td>
<td>700</td>
<td>300</td>
<td>210,000</td>
<td>0</td>
<td>0</td>
<td>240</td>
<td>168,000</td>
</tr>
<tr>
<td>9</td>
<td>Fuel</td>
<td>4,500</td>
<td>690</td>
<td>3,105,000</td>
<td>0</td>
<td>0</td>
<td>430</td>
<td>1,935,000</td>
</tr>
<tr>
<td>10</td>
<td>Batteries</td>
<td>18,000</td>
<td>733</td>
<td>13,194,000</td>
<td>0</td>
<td>0</td>
<td>120</td>
<td>2,160,000</td>
</tr>
<tr>
<td>11</td>
<td>Passengers</td>
<td>16,800</td>
<td>680</td>
<td>11,424,000</td>
<td>0</td>
<td>0</td>
<td>200</td>
<td>3,360,000</td>
</tr>
<tr>
<td>12</td>
<td>Luggage</td>
<td>3,000</td>
<td>680</td>
<td>2,040,000</td>
<td>0</td>
<td>0</td>
<td>200</td>
<td>600,000</td>
</tr>
</tbody>
</table>
Figure 61. c.g. for various aircraft components
Using the listed c.g. locations for the various components and their individual weights, the c.g. excursion diagram was produced using Excel. The most forward c.g. is located at 46.91 ft. and the most aft c.g. is located at 66.08 ft. Although they have a very wide spread, AAA will be used to check these calculations, and then after the landing gear placement, the weight and balance will be performed again.
Using the 737-200 airplane for comparison, the weight fractions were calculated for the various components. The figure below illustrates the results from AAA.

<table>
<thead>
<tr>
<th>Selected Airplanes Weight Fractions</th>
</tr>
</thead>
<tbody>
<tr>
<td>#</td>
</tr>
<tr>
<td>1</td>
</tr>
</tbody>
</table>

Output Parameters:

- \(W_{\text{gross}}\): 1.000
- \(W_{\text{structure}}\): 0.278
- \(W_{\text{fuselage}}\): 0.071
- \(W_{\text{engine}}\): 0.124
- \(W_{\text{wing}}\): 0.521
- \(W_{\text{gear}}\): 0.002
- \(W_{\text{I}}\): 0.024
- \(W_{\text{II}}\): 0.105
- \(W_{\text{III}}\): 0.012
- \(W_{\text{IV}}\): 0.018

Figure 63. c.g. excursion plot produced using Microsoft Excel

Figure 64. Weight fractions referenced from the 737-200 from AAA

Using the empty weights, the c.g. positions for each of the empty weight components can be determined, as well as the c.g. of the structure as a whole. These results, from AAA, are summarized as follows.
Design of a Hybrid Transport Aircraft

Figure 65. c.g. locations produced using AAA

Using the outputted parameters as inputs, the c.g. of the HTA along with the c.g. locations for each component.

Figure 66. c.g. location for MTOW aircraft

From here, the c.g. excursion diagram for the x and z directions can be plotted. The following information was used to plot the c.g. excursion diagram.
Using the loading table and the calculated c.g. values, the x c.g. excursion plot illustrates the possible c.g. locations depending on the loading scenario.
The plot illustrates a variation in c.g. with the most forward c.g. being located at 22.9 ft and the most aft c.g. being located at 35.7 ft. Note that the c.g. is left of the main landing gear, which indicates that the HTA will not tip over with the current loading and unloading scenarios. For reference, the z-direction c.g. excursion diagram has also been plotted. Note that for the loading and unloading scenarios, the cargo represents the batteries.
Design of a Hybrid Transport Aircraft

**Figure 69.** Loading Table from AAA for z-direction c.g. excursion diagram

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight</th>
<th>Z&lt;sub&gt;CG&lt;/sub&gt;</th>
<th>Load</th>
<th>Unload</th>
</tr>
</thead>
<tbody>
<tr>
<td>Empty Weight</td>
<td>39593.8</td>
<td>7.44</td>
<td>1</td>
<td>13</td>
</tr>
<tr>
<td>Crew</td>
<td>760.0</td>
<td>20.00</td>
<td>6</td>
<td>8</td>
</tr>
<tr>
<td>Trapped Fuel and Oil</td>
<td>467.6</td>
<td>17.92</td>
<td>2</td>
<td>12</td>
</tr>
<tr>
<td>Mission Fuel Group 1</td>
<td>15018.3</td>
<td>17.08</td>
<td>3</td>
<td>1</td>
</tr>
<tr>
<td>Mission Fuel Group 2</td>
<td>0.0</td>
<td>0.00</td>
<td>4</td>
<td>2</td>
</tr>
<tr>
<td>Passenger Group 1</td>
<td>4200.0</td>
<td>16.57</td>
<td>7</td>
<td>3</td>
</tr>
<tr>
<td>Passenger Group 2</td>
<td>4200.0</td>
<td>16.57</td>
<td>8</td>
<td>4</td>
</tr>
<tr>
<td>Passenger Group 3</td>
<td>4200.0</td>
<td>16.57</td>
<td>9</td>
<td>5</td>
</tr>
<tr>
<td>Passenger Group 4</td>
<td>4200.0</td>
<td>16.57</td>
<td>10</td>
<td>6</td>
</tr>
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<td>11</td>
<td>7</td>
</tr>
<tr>
<td>Batteries</td>
<td>18000.0</td>
<td>10.00</td>
<td>5</td>
<td>9</td>
</tr>
<tr>
<td>Military Load Group 1</td>
<td>0.0</td>
<td>0.00</td>
<td>12</td>
<td>10</td>
</tr>
<tr>
<td>Military Load Group 2</td>
<td>0.0</td>
<td>0.00</td>
<td>13</td>
<td>11</td>
</tr>
</tbody>
</table>

**Figure 70.** c.g. excursion diagram (z-direction)
8.3. Landing Gear Design

For the landing gear design, the following characteristics will be determined: number and size of tires, length and diameter of struts, preliminary disposition, and retraction feasibility. The HTA will have retractable tricycle landing gear. Preliminary decisions on placement of the wheels and struts will be made, and the placements will be checked to see that they meet the tip over criteria and ground clearance criteria.

The main landing gear must be behind the aft most c.g. Based on the c.g. excursion diagram, the most aft c.g. is located at 39.63 ft. By constructing a line parallel ot the sweep of the fuselage and a 7-15 degree angle to find the optimal ground clearance, the intersection of these two is the location of the main landing gear. This location, at 47.33 ft, which is the same as the location of the Boeing 737-100 model, is aft of the c.g. Thus, it meets the tip-over criteria.

To meet the ground clearance criteria, the main gear struts will be placed under the fuselage. The selection of the strut length will impact the weight of the landing gear, ground clearance of the aircraft, the tip-over characteristics, and stability during ground operation.

To determine the maximum static load per strut, the following equations will be used:

\[ P_n = \frac{W_{TO}l_m}{l_m + l_n} \]  \hspace{1cm} (51)

\[ P_m = \frac{(W_{TO}l_n)}{n_s(l_m + l_n)} \]  \hspace{1cm} (52)

Note that \( P_n \) and \( P_m \), are the maximum nose wheel strut loading and main gear strut loading, and \( l_m \) and \( l_n \), are the distance from the c.g. to the main landing gear and nose gear, respectively. For the HTA, the maximum load per strut for the main landing gear is 32,833 lbs, and the maximum load per strut for the nose gear is 27,852 lbs.

There will be two nose wheels and four main wheels. The nose wheel tires have a diameter of 24 inches and a wheel tread width of 7.75 inches. The main wheel tires have a diameter of 40 inches and a wheel tread width of 14.5 inches (Brady, 1999). Boeing oleo pneumatic shock
aborbers provide shock absorption. One main gear strut and one nose gear strut dampen the vertical oscillations felt upon landing.

*Figure 71. Landing gear and strut placement*

*Figure 72. Nose gear retraction*
**Figure 73.** Stick Diagram for main gear retraction

**Figure 74.** Illustration of the two nose tires and four main tires for the landing gear.

**Figure 75.** Landing Gear retracted into fuselage
8.4. Weight and Balance

Since the landing gear placement, at 47 feet from the nose, is aft of the most aft c.g., the weight and balance does not need to be repeated. The location of the main landing gear correlates with the location of the main landing gear in the 737-100 model.

8.5. Conclusion

Compared to the 737-100 model, the c.g. excursion diagram demonstrated the change in c.g. location due to the addition of the batteries. The location of the batteries did not change the location of the main landing gear, and, as a result, the HTA will not tip over in any loading scenario. The landing gear can support the weight of the plane, with the added weight due to the batteries, and the landing gear retracts directly into the fuselage.

9. Stability and Control Analysis

9.1. Introduction

The stability and control analysis will be performed by following Roskam’s steps in Airplane Design Part II. This will allow the determination of the stability and control characteristics of the HTA. These characteristics include the static longitudinal stability, static directional stability, and minimum control speed with one engine out (Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System, 2011)

Stability is the tendency of an object to return to equilibrium when displaced (Crawford, 2009). Static stability involves an object’s initial tendency to return to equilibrium upon initial displacement. For aircraft, this requires a balance because the greater an aircraft’s static stability, the greater its resistance in maneuvering. The center of gravity, or c.g., strongly determines the longitudinal static stability. Moving the c.g. aft reduces static stability and increases maneuverability.
Aircraft can have inherent stability, in which the stability is built into the aircraft’s design, or de facto stability, where control systems provide feedback and provide stability requirements. Some aircraft have relaxed stability. This type of stability involves the aircraft changing its attitude and bank angle on its own. This, along with stability augmentation devices, allows for better cruise performance and better maneuverability at high altitudes. Many jet transport aircraft have relaxed stability. The HTA will also feature relaxed stability.

9.2. Static Longitudinal Stability

The longitudinal X-plot illustrates the rate at which the c.g. moves aft as a function of the horizontal tail area and the rate at which the aerodynamic center, or a.c., moves aft as a function of the horizontal tail area. This will be plotted for the HTA, which features a conventional configuration.

The a.c. leg of the plot will be calculated using the following equations:

\[
\bar{x}_{\text{aC}} = \left[ \frac{\bar{x}_{\text{acwf}} + \left( C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \frac{C_{th}}{2} \right) \bar{x}_{\text{ach}} - C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \bar{x}_{\text{acw}} + C_{\text{LaC}} \bar{x}_{\text{acw}}}{C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \frac{C_{th}}{2} \bar{x}_{\text{ach}} - C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \bar{x}_{\text{acw}} + C_{\text{LaC}} \bar{x}_{\text{acw}}} \right]
\]

(53)

\[
F = \frac{\left[ 1 + \left( C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \frac{C_{th}}{2} \bar{x}_{\text{ach}} - C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \bar{x}_{\text{acw}} + C_{\text{LaC}} \bar{x}_{\text{acw}} \right) \right]}{C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \frac{C_{th}}{2} \bar{x}_{\text{ach}} - C_{\text{LaC}} \left( \frac{d\alpha}{da} \right) \bar{x}_{\text{acw}} + C_{\text{LaC}} \bar{x}_{\text{acw}}}
\]

(54)

The figure below illustrates how to find the geometric quantities to perform these calculations.

(Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System, 2011)
The longitudinal x plot illustrates that as the tail plane area is increased, the c.g. moves aft, and the a.c. also moves aft, but more slowly. The 10% difference between the a.c. and c.g. location is the optimal area of the horizontal stabilizer.

The 10% static margin yields a horizontal stabilizer area of 380–401 ft². These results match the results from the wing design, where the horizontal stabilizer area was sized to 401 ft².
This yields a slightly larger wing area than the 737-100, but this is likely due to the extra battery weight being suspended below the wings.

The HTA has a level of instability of $0.085\bar{e}_w$, which has been chosen based on the instability levels of comparable jet transport. Using the aft c.g. leg of the longitudinal X-plot, the longitudinal stability augmentation system must generate a change in static margin of $\Delta SM = 0.085 + 0.05 = 0.135$, where the minimum static margin of $X_{cg} - X_{ac} = 0.05$ or 5%. The empennage area follows from these calculations. The original horizontal tail area of 401 ft$^2$ will be compared with the horizontal tail area obtained from the longitudinal X-plot. Since they are the same, 401 ft$^2$ will remain the design size.

### 9.3. Static Directional Stability

The directional X-plot illustrates the change in directional stability for various vertical tail areas.

*Figure 78. HTA Directional X-Plot*

The desired de facto level of directional instability is $C_{n_{\beta}} = 0.0010$. From the plot, $C_{n_{\beta}} = -0.001$ and there is a difference of 0.002. The sideslip feedback system must support
this difference. To compute the required sideslip to rudder feedback gain, the following equations will be used.

\[
k_B = \frac{\Delta C_{n\beta}}{C_n \delta_r}, \text{ with } k_B < 5 \text{ deg/deg} \tag{55}\]

\[
\Delta C_{n\beta} = 0.0010 - C_{n\beta} \tag{56}
\]

As a result, \( k_B = \frac{\Delta C_{n\beta}}{C_n \delta_r} = \frac{0.0010 - 0.002}{-0.0012} = 0.833 \text{ deg/deg} \). This is acceptable, and the vertical tail is not critical for the HTA.

9.4. Minimum Control Speed with One Engine Inoperative

The critical one engine out yawing moment can be determined from the following equation.

\[
N_{t_{crit}} = T_{T0e} Y_t \tag{57}
\]

In this equation \( Y_t \) is the lateral thrust moment arm of the most critical engine, which equals 17.5 ft. \( T_{T0e} \) is the thrust required by the engine at takeoff. The geometry for these calculations is illustrated in the figure below.

Figure 79. Geometry for one engine out \( V_{mc} \) calculations
Using the geometry and the thrust at takeoff, \( N_{t_{crit}} = T_{ROe} Y_t = 21,000 lb \cdot 17.5 ft = 367,500 \text{ ft lbs} \). For a jet transport with a low bypass ratio, \( N_D = 0.15N_{t_{crit}} \), and thus \( N_D = 55,125 \text{ ft lbs} \). Therefore, the critical engine out yawing moment is 55,125 ft lbs.

The following equations can be used to calculate the maximum allowable \( V_{mc} \) and the rudder deflection required to hold the engine out condition at \( V_{mc} \). The rudder deflection angle should not exceed 25 degrees. If it does, the vertical tail needs to be resized or the rudder needs to be resized.

\[
V_{mc} = 1.2 V_s \tag{58}
\]
\[
\delta_r = (N_D + N_{t_{crit}})/\bar{q}_{mc} S b C_{n_{\delta r}} \tag{59}
\]

Using the lowest stall speed, which is the stall speed at landing, \( V_{mc} \) is calculated to be \( 1.2 V_{s_L} = 1.2(54) = 64.8 \text{ kts} \). From the vertical tail and rudder geometry, the control power derivative is equal to \( C_{n_{\delta r}} = -0.245 \text{ rad}^{-1} \). Thus, at \( V_{mc} = 64.8 \text{ kts} \), the rudder deflection angle is calculated to be

\[
\delta_r = \frac{T_{ROe} Y_t}{-\bar{q} S b C_{n_{\delta r}}} = \frac{(55,125 \text{ ft lbs} + 367,500 \text{ ft lbs})(.5(7.38 \times 10^{-4})(109.37 ft/s)^2(250 ft^2)(93.011 ft)\left(-\frac{0.0012}{\text{deg}}\right)}{43.6^\circ}. \]

This is a large angle, and the size of the vertical tail should be increased and the calculation performed again.

9.5. Empennage Design-Weight and Balance- Landing Gear Design- Longitudinal Static Stability and Control Check

The horizontal stabilizer area meets the 10% static margin difference, and the horizontal stabilizer does not need to be resized. The vertical stabilizer needs to increase in order to provide a smaller rudder deflection angle.
9.7. Conclusion

Overall, the sizing of the HTA is still comparable to the 737-100 model. The horizontal stabilizer was increased to 401 ft², as determined in the wing sizing chapter, and this sizing will be maintained based on the longitudinal X-plot.

The critical one engine out yawing moment has been determined to be 55,125 ft lbs, with a minimum control speed of 64.8 kts.

Rudder deflection should not exceed 25 degrees, but was calculated to be 43.6°, which necessitates an increase in size of the vertical stabilizer. This will be explored further during the Class II sizing.

10. Drag Polar Estimation

10.1. Introduction

A drag polar is the relationship between the airplane’s lift and its drag, where the lift coefficient depends on the drag coefficient. The results of the HTA’s comparison of its lift and drag coefficients will be examined and displayed in a plot.

To perform the drag polar estimation, Roskam’s method will be utilized. This involves eight steps. Initially, the wetted area of the airplane components will be estimated. The aircraft’s components will be analyzed individually. These include the fuselage, wings, empennage, and engine and battery nacelles. Once these have been computed, the equivalent parasite area will be calculated. From there, the compressibility drag increment will be found along with the flap drag increment, and landing gear drag increment. The drag polars for cruise, takeoff, and landing will be constructed, and the L/D critical values will be determined from the drag polars.

10.2. Airplane Zero Lift Drag

The following equations will be used to compute the wetted areas for each component.
Design of a Hybrid Transport Aircraft

\[ S_{\text{wet,plf}} = 2S_{\text{exp,plf}} \left( 1 + 0.25 \left( \frac{l_c}{c} \right)_r \frac{(1 + \tau \lambda)}{(1 + \lambda)} \right). \]  
\[ \tau = \left( \frac{l_c}{c} \right)_r \left( \frac{l_c}{c} \right)_t \]  
\[ \lambda = c_t/c_R \]  
\[ S_{\text{wet,fus}} = \pi D_f l_f \left( 1 - \frac{2}{\lambda_f} \right)^\frac{3}{2} \left( 1 + 1/\lambda_f^2 \right) \]  
\[ \lambda_f = l_f/D_f \]  
\[ S_{\text{wet,fan}} = l_n D_n \left\{ 2 + \frac{0.35 l_1}{l_n} + \frac{0.8 l_1 D_{h} l_n}{l_n D_n} + \frac{1.15(1 - \frac{l_n}{l_n}) D_{ef}}{D_n} \right\} \]  
\[ S_{\text{wet,gen}} = \pi l_g D_g \left[ 1 - \left( \frac{l_c}{3} \right) \left( 1 - \frac{D_{eg}}{D_g} \right) \left( 1 - 0.18 \left( \frac{D_g}{l_g} \right)^{5/3} \right) \right] \]  
\[ S_{\text{wet,plug}} = 0.7 \pi l_p D_p \]  

It’s important to note that the wings, empennage, and nacelles usually intersect the fuselage or a nacelle, so the intersection of the wetted area will be subtract. In order to estimate the wetted area of the batteries, the calculated wetted area of the engine nacelles will be multiplied by a factor of 1.5.

The wetted areas are as follows:

<table>
<thead>
<tr>
<th>Component</th>
<th>Wetted Area (ft²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>2,367</td>
</tr>
<tr>
<td>Intersection of wing and fuselage</td>
<td>-200</td>
</tr>
<tr>
<td>Vertical Tail</td>
<td>463</td>
</tr>
<tr>
<td>Horizontal Tail</td>
<td>642</td>
</tr>
<tr>
<td>Engine Nacelles</td>
<td>455</td>
</tr>
<tr>
<td>Battery Nacelles</td>
<td>455</td>
</tr>
<tr>
<td>Fuselage</td>
<td>4,000</td>
</tr>
<tr>
<td>TOTAL WETTED AREA</td>
<td>8,182</td>
</tr>
</tbody>
</table>

The wetted area compares fairly with the wetted area of a jet transport that weights approximately 90,000 lbs and has a wetted area of 6,000 ft², as illustrated in the graph below (Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the...
Propulsion System, 2011). The HTA has a larger wetted area due to the battery nacelles. This added area may impact the cruise L/D and the corresponding fuel consumption.

*Figure 80. Maximum takeoff weight vs. wetted area for jet transports*

Using the plot below, the equivalent parasite area of the HTA is approximately 22 ft$^2$. 

```
MAXIMUM TAKE-OFF WEIGHT ~ W_{TO} ~ LBS

WETTED AREA ~ S_{WET} ~ FT^2

7 TRANSPORT JETS

MAXIMUM TAKE-OFF WEIGHT ~ W_{TO} ~ LBS

10^3 10^4 10^5 10^6

1 2 3 4 5 6 7 8 9 10

4 3 2 1

2 3 4 5 6 7 8 9 10

10^5

10^4

10^3

10^2
```
10.3. Low Speed Drag Increments and Compressibility Drag

The clean zero lift drag coefficient at low speed is \( C_{D_0} = \frac{f}{s} = \frac{22}{1098} = 0.02 \). Using the figure below, the compressibility drag increment is about 0.0005. Due to this low value the cruise drag does not need to be re-evaluated. The cruise value of the zero lift drag coefficient can be summarized as \( C_{D_0} = 0.0005 + 0.02 = 0.0205 \). The minimum lift to drag ratio can be calculated and then compared to the previously determined L/D.

Originally, \( C_{D_0} \) was estimated to be:
Table 36. $C_D_0$ summary

<table>
<thead>
<tr>
<th>Configuration</th>
<th>$C_D_0$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Clean</td>
<td>0.014</td>
</tr>
<tr>
<td>Takeoff flaps</td>
<td>0.03</td>
</tr>
<tr>
<td>Landing flaps</td>
<td>0.08</td>
</tr>
<tr>
<td>Gear down</td>
<td>0.02</td>
</tr>
</tbody>
</table>

Using the clean $C_D_0$ value and the newly calculated $C_D_0 = 0.0205$, the $(L/D)_{\text{max}}$ values are equal to 18.27 and 18.05, respectively. Based on the sensitivity calculation, for each increase in $L/D$, there is a decrease of 856 lbs. Thus, for a change of 0.224, the weight will increase 192 lbs, which is not substantial enough to change the structure.

Figure 82. Typical Compressibility Drag Behavior

10.3.1. High Lift Device Drag Increments for Takeoff and Landing Gear Drag

Table 37. First Estimates for Drag with flaps and gear down

<table>
<thead>
<tr>
<th>Configuration</th>
<th>$\Delta C_D_0$</th>
<th>$e$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Clean</td>
<td>0</td>
<td>0.80 - 0.85</td>
</tr>
<tr>
<td>Takeoff flaps</td>
<td>0.010 - 0.020</td>
<td>0.75 - 0.80</td>
</tr>
<tr>
<td>Landing Flaps</td>
<td>0.055 - 0.075</td>
<td>0.70 - 0.75</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>0.015 - 0.025</td>
<td>no effect</td>
</tr>
</tbody>
</table>
The values in table 2 will be used to produce the drag polar plot based on the change in $C_{D_0}$ for various configurations, including clean, takeoff flaps, landing flaps, and landing gear.

10.5. Airplane Drag Polars

![Figure 83. Low speed polars for transport](image)

![Figure 84. L/D Ratios](image)

10.6. Discussion

The HTA will not be resized. Although it is heavier than anticipated, it is within 10%, and was expected, due to the battery weight. A more thorough estimation would probably include subtracting more overlapping wetted areas, as well as more accurately sizing the battery nacelles.
10.7. Conclusion

The HTA is ready to commence Class II sizing following an analysis of the economic, safety, and environmental tradeoffs.

11. Environmental and Economic Tradeoffs

11.1. Important Design Parameters

The HTA three view drawing that reflections the iterations of the design is illustrated in

![Figure 85. 3-view of HTA](image-url)
The most important feature of this aircraft is the implementation of lithium ion batteries for battery powered cruise, loiter, descent, and landing. These batteries changed the weight analysis and overall sizing of the aircraft, but many of the features, such as the fuselage design, remained the same as the original 737-100. The following will summarize economic and environmental tradeoffs, as well as safety tradeoffs for the HTA.

11.2. Recommendations

To further support this design concept, more research needs to be done for the overall battery weight and volume sizing. The sizing summarized in the reports reflected conceptual designs and sizings that have been done for both similar sized aircraft, such as Boeing’s conceptual SUGAR designs, and small aircraft, through the work of Riboldi. Additionally, more research could be done with lithium ion batteries to determine if the battery specific energy could be increased further. In terms of design, the vertical stabilizer size needs to increase as well. This is to support a lower rudder deflection angle.

11.2.2. Environmental/Economic Tradeoffs

12. Preliminary Design Sequence II

This chapter commences the Class II sizing of the HTA. The systems that have the largest impact on the HTA include the propulsion system, specifically engines and the fuel and battery power to support the system; aerodynamic control systems, including the ailerons, elevator, and rudder; and aircraft structural system, which include the wings, fuselage, horizontal tail, and vertical tail. These systems have a significant impact on the empty weight of the aircraft, and will be thoroughly analyzed during the weight and balance analysis. Additionally, a ghost view of the aircraft, presented in Figure 86, illustrates the aircraft systems and their current locations.
Initially, a V-n diagram will be constructed, and then the Class II weight and balance analysis will be performed. From there, the Class II stability and control analysis will be conducted, followed by the Class II drag polar analysis. The power computation and installation will follow, analyzing power for the aircraft along with factors that change the power output, such as altitude. Finally, a cost analysis will be performed to predict the acquisition cost of the airplane. This will be performed based on statistical data and equations for cost.

13. V-n Diagram

A V-n diagram is a graph of the velocity vs. load factor, which illustrates the aircraft performance limits. These limits include the design ultimate load factors and the speed to which each of the airplane’s parts have been designed (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). This will be used with the Class II weight estimation and will be considering the case where flaps are up.
For the calculations, the following nomenclature will be used: $V_C$ = design cruise speed, $V_D$ = design diving speed, $V_M$ = design maneuvering speed, $V_S$ = $+1g$ stall speed or minimum speed at which airplane is controllable, and $V_B$ = design speed for maximum gust intensity.

13.1. Calculation of $V_S$

To determine the minimum controllable speed of the FAR 25 aircraft, the following equation will be used: (Roskam, Airplane Design Part V: Component Weight Estimation, 2003)

$$V_S = \left( \frac{2GW}{S} \right) \left( \frac{1}{\rho C_{N_{\text{max}}}} \right)^{1/2} \quad (68)$$

Note that GW is the flight design gross weight, measured in lbs; S is the wing area in square feet; the air density is measured in slugs/cubic feet; and $C_{N_{\text{max}}}$ is defined as:

$$C_{N_{\text{max}}} = \left( C_{L_{\text{max}}} \right)^2 + \left( C_{D_{\text{at } C_{L_{\text{max}}}}} \right)^2 \right)^{1/2} \quad (69)$$

Another way to calculate $C_{N_{\text{max}}}$ is using the following:

$$C_{N_{\text{max}}} = 1.1C_{L_{\text{max}}} \quad (70)$$

Solving these equations gives $C_{N_{\text{max}}} = 1.56$ and $V_S = 129.8$ kts.

13.2. Calculation of $V_C$

The design cruise speed must be greater than the design speed for maximum gust intensity in the event that there are severe atmospheric turbulences that could cause speed increases. To determine $V_B$, consider the intersection of the $C_{N_{\text{max}}}$ line and the $V_B$ gust line. The value of $V_B$ may not be less than the value at the intersection. From there, using the equation below, $V_C = 290$ kts.

$$V_C \geq V_B + 43 \text{ kts} \quad (71)$$
13.3. Calculation of $V_D$

To calculate the design driving speed, the following equation will be used:

$$V_D \geq 1.25V_C$$  \hspace{1cm} (72)

As a result, $V_D \geq 368.75$ kts.

13.4. Calculation of $V_M$

The design maneuvering speed is given by the following condition:

$$V_M \geq V_s (n_{lim})^{\frac{1}{2}}$$  \hspace{1cm} (73)

Therefore, $V_M = 205.23$ kts. The limit maneuvering load factor at $V_C$ depends on the more critical requirement of either the positive limit maneuvering load factor or the gust load factor lines. The positive maneuvering load factor, $n_{lim_{pos}}$. Where the following equation determines the positive limit maneuvering load factor.

$$n_{lim_{pos}} \geq 2.1 + \left\{ \frac{24,000}{W+10,000} \right\}$$  \hspace{1cm} (74)

The only exception to this is that the limit maneuvering load factor must be greater than or equal to 2.5 at all times, but not greater than 3.8 at the takeoff weight. Thus, although $n_{lim_{pos}}$ is equal to 2.4, following the exception, $n_{lim_{pos}} = 2.5$. The negative maneuvering load factor has two specifications. This value is greater than or equal to -1 up to the design cruise velocity and from the cruise speed design to the the design diving speed, the negative maneuvering load varies linearly.

To determine the negative stall speed line, the following equations will be used:

$$V_{S_{neg}} = \left\{ \frac{2(GW)}{\mu C_{N_{max_{neg}}}} \right\}^{1/2}$$  \hspace{1cm} (75)

$$C_{N_{max_{neg}}} = \left\{ (C_{l_{max_{neg}}})^2 + (C_D_{at_{C_{l_{max_{neg}}}}})^2 \right\}^{1/2}$$  \hspace{1cm} (76)
Design of a Hybrid Transport Aircraft

\[ C_{N_{\text{max}_{\text{neg}}}} = 1.1 C_{\ell_{\text{max}_{\text{neg}}}} \]  
(77)

It will be assumed that \( C_{\ell_{\text{max}_{\text{neg}}}} = -1.0 \), from similar transport aircraft. As a result, \( C_{\ell_{\text{max}_{\text{neg}}}} = -1.1 \) and \( V_{s_{\text{neg}}} = 154 \) kts.

The design maneuvering speed depends on the construction of the gust load factor lines if they are more critical. Thus the derived gust velocities will be determined. To plot the gust lines, the following equations will be used (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[ n_{\text{lim}} = 1 + \left( K_g U_{de} V C_{q_a} \right) / (498 G W / S) \]  
(78)

\[ K_g = 0.88 \mu_g / (5.3 + \mu_g) \]  
(79)

\[ \mu_g = 2 \left( \frac{G W}{S} \right) / (\rho (m g c) g C_{q_a}) \]  
(80)

In these equations, \( K_g \) is the gust alleviation factor, and the gust load factor for the lines is defined using \( n_{\text{lim}} \). Note that \( V \) is the true airspeed, and that is the variable. Since \( n_{\text{lim}} \) also depends on \( U_{de} \), those values, for each of the gust lines, for \( V_B, V_C, \) and \( V_D \), can be determined using the equations below (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[ V_{B_{de}} = \begin{cases} 66 \text{ fps for } 0 \leq h \leq 20,000 \text{ ft} \\ 84.67 - 0.000933h \text{ for } 20,000 \leq h \leq 50,000 \text{ ft} \end{cases} \]  
(81)

\[ V_{C_{de}} = \begin{cases} 50 \text{ fps for } 0 \leq h \leq 20,000 \text{ ft} \\ 66.67 - 0.000833h \text{ for } 20,000 \leq h \leq 50,000 \text{ ft} \end{cases} \]  
(82)

\[ V_{D_{de}} = \begin{cases} 25 \text{ fps for } 0 \leq h \leq 20,000 \text{ ft} \\ 33.34 - 0.000417h \text{ for } 20,000 \leq h \leq 50,000 \text{ ft} \end{cases} \]  
(83)

Assuming a cruise height of 35,000 feet, the gust lines in Figure 88, below, were graphed based on the equations:

\[ V_B = 1 + 0.004835V \]  
(84)

\[ V_C = 1 + 0.003487V \]  
(85)
The negative gust lines had negative slopes, but still started at 1.

13.5. V-n Maneuver Diagram and V-n Gust Diagram

The following graphs summarize the design limits, by illustrating the V-n Maneuver Diagram and the V-n Gust Diagram for the HTA.

![V-n Maneuver Diagram]

*Figure 87. V-n Maneuver Diagram*

![V-n Gust Diagram]

*Figure 88. V-n Gust Diagram*
According to the diagrams in Figure 87 and Figure 88, the positive load factor that can be safely achieved at varying airspeeds ranges from 0 to 2.5, while the negative load factor cannot go below -1. Any velocity and load factor combination that lies outside of the line would result in a stall; whereas, any point inside the lines is within the flight envelope, and is possible. These diagrams were constructed assuming symmetrical loading. This chart has been plotted for 35,000 feet, but can be plotted at various altitudes to illustrate the loading factor achievable. The gust lines fall within the maneuvering V-n diagram.

14. Class II Weight and Balance

From the Class I analysis, the following weights are already known: payload, crew, fuel, batteries, trapped fuel and oil, and engine weight. The remaining weights will need to be estimated. The weight of the structure consists of the wing, fowler flaps, empennage, fuselage, nacelles, and landing gear. The weight of the powerplant consists of the engines, fuel system, propulsion system, accessory drives, and thrust reversers. The fixed equipment weight consists of flight controls, electrical system, instrumentation, avionics, electronics, airconditioning, pressurization syste, de-icing system, oxygen, APU, furnishings, baggage and cargo handling, operational items, and paint. Equations used in Roskam’s fifth volume will be used to compute the weights of the structure, power, and fixed equipment. From there, the Class II empty weight will be compared to the Class I empty weight. Iterations will be performed if there is a discrepancy between the two weights.

14.1. Class I Weights

The following table summarizes the known weight values from the Class I sizing.
Table 38. Class I Weights

<table>
<thead>
<tr>
<th>Item</th>
<th>Nomenclature</th>
<th>Weight (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>payload</td>
<td>( W_{pl} )</td>
<td>19,680</td>
</tr>
<tr>
<td>Crew</td>
<td>( W_{crew} )</td>
<td>820</td>
</tr>
<tr>
<td>Fuel</td>
<td>( W_f )</td>
<td>4,140</td>
</tr>
<tr>
<td>Trapped Fuel and Oil</td>
<td>( W_{tfo} )</td>
<td>150</td>
</tr>
<tr>
<td>Batteries</td>
<td>( W_{bat} )</td>
<td>10,300</td>
</tr>
</tbody>
</table>

14.2. Weight Estimate Calculations

14.2.1. Structural Weight

The structural weight, \( W_{struct} \), consists of the wing, \( W_w \), fowler flaps, empennage, \( W_{emp} \), fuselage, \( W_{fus} \), nacelles, \( W_n \), and landing gear, \( W_g \). Both the GD Method and the Torenbeck method will be used to compute the weight of the wing. Both methods, which can be used for transport aircraft, consider the weight of normal high lift devices and ailerons.

14.2.1.1. Wing Weight

The GD method will use the equation, below, where the maximum Mach number at sealevel should be between 0.4 to 0.8 and the maximum wing thickness ratio should range from 0.08 to 0.15 (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). The HTA cruises at Mach 0.75 and has a thickness ratio of 0.124. As a result, the GD Method can be applied for estimation and the equation can be used.

\[
W_w = \left[ 0.00428 \left( S^{0.48} \right) \left( A \right) \left( M_H \right)^{0.43} \left( W_{to} n_{ult} \right)^{0.84} \lambda^{0.14} \right] / \left[ 100 \left( \frac{L}{c} \right)_m \right]^{0.76} \left( \frac{\alpha \Delta A}{\theta} \right)^{1.54}\]  

(87)

Additionally, the aircraft meets specifications for the Torenbeek sizing, as it is a transport with a take-off weight greater than 12,500 lbs. The following equation presents Torenbeek’s method (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).
\[ W_w = 0.0017W_{MZF} \left( \frac{b}{\cos \left( \frac{A_1}{\bar{z}} \right)} \right)^{0.75} \left( 1 + \left( \frac{6.3 \cos \left( \frac{A_1}{\bar{z}} \right)}{b} \right)^{0.5} \left( n_{ult}^{0.055} \right) \left( \frac{bs}{t_rW_{MZF}\cos A_1} \right)^{0.3} \right) \] (88)

The GD Method gives an estimation of 4,872 lbs, and the Torenbeek Method gives an estimation of 8,038. The Class I sizing provided an estimate of 7,636 lbs. Averaging the three weights gives 6,849 lbs. However, considerations must be made for the number of engines and Fowler flaps. With 2 suspended engines and 2 suspended battery nacelles, the weight will initially be reduced by 10%, and with the Fowler flaps, the weight is increased by 2%. The resulting Class II estimate of the wing weight is 6,287 lbs.

14.2.1.2. Empennage Weight

The Empennage consists of the sum of the horizontal tail and vertical tail. For commercial transport jets, like the HTA, the GD Method and the Torenbeek Method will be applied to estimate. From there, comparisons will be made to the Class I estimate and a final Class II estimate will be established.

Using the GD Method, the horizontal tail and vertical tail can be estimated using the following two equations (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[ W_h = 0.0034 \left( \left( W_{to n_{ult}} \right)^{0.813} \left( S_h \right)^{0.584} \left( \frac{b h}{t_r h} \right)^{0.033} \left( \frac{m g c}{l_h} \right)^{0.28} \right)^{0.915} \] (89)

\[ W_v = 0.19 \left( 1 + \frac{z_n}{b_v} \right)^{0.5} \left( W_{to n_{ult}} \right)^{0.363} \left( S_v \right)^{1.089} \left( M_H \right)^{0.601} \left( l_v \right)^{-0.726} \left( 1 + \frac{S_r}{S_v} \right)^{0.217} \left( A_v \right)^{0.337} \left( 1 + \frac{\lambda_v}{\bar{z}} \right)^{-0.484} \left( \cos A_1 \right)^{1.014} \] (90)

Torenbeek’s Method involves the following equations (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).
Design of a Hybrid Transport Aircraft

\[ W_h = K_h S_h (3.81 (S_h)^{0.2} V_D) / (1000 \cos \frac{A_1}{z_h})^{\frac{1}{2}} - 0.287 \]  

(91)

\[ W_v = K_v S_v (3.81 (S_v)^{0.2} V_D) / (1000 \cos \frac{A_1}{z_v})^{\frac{1}{2}} - 0.287 \]  

(92)

For fixed incidence stabilizers, \( K_h = 1 \), and for fuselage mounted horizontal tails, \( K_v = 1 \).

The GD Method yields a horizontal tail weight of 831.649 lbs; while the Torenbeek Method produces a horizontal tail weight of 1,169 lbs. The GD Method provides a vertical tail area of 2,797 lbs, and the Torenbeek Method calculated 1,093 lbs. The total empennage weight from GD is 3,628 lbs, and the total empennage weight from Torenbeek is 2,263 lbs. The Class I estimate had an empennage weight of 1,992 lbs. Taking the average of the three yields 2,627 lbs for the Class II estimate of the empennage.

**14.2.1.3. Fuselage Weight**

For commercial transport aircraft, the fuselage weight can be calculated using the GD Method and the Torenbeek Method. Again, both calculations will be performed to assess the best estimate for Class II fuselage weight.

The GD Method provides the following equation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[ W_{fus} = 10.43 (K_{int})^{1.42} \left( \frac{\bar{q}}{100} \right)^{0.283} \left( \frac{W_{TTO}}{1000} \right)^{0.95} \left( \frac{L_{fus}}{h_{fus}} \right)^{0.71} \]  

(93)

In the above equation, \( K_{int} \) is the factor that takes into account whether there are inlets on the fuselage for buried engines. Since the engines are suspended and not buried, \( K_{int} = 1 \), and \( \bar{q} \) is the design dive dynamic pressure in psf.

The Torenbeek Method, which applies to jet transport aircraft with dive speeds above 250 kts, provides the following equation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).
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\[ W_{fus} = 0.021 K_f \left( \frac{V_D l_h}{w_{fus} + h_{fus}} \right)^{1/2} (S_{f,g})^{1.2} \]  

(94)

\( S_{f,g} \) is the fuselage gross shell area in feet. \( K_f = 1.08 \) for a pressurized fuselage. The weight of the fuselage is a function of the dive speed, width and height of the fuselage, and the distance from the wing root quarter cord to the the tail root quarter cord in feet.

Using the GD Method, the fuselage weight is 4024 lbs. The Torenbeek Method calculates the fuselage weight to be 7425 lbs, and the Class I estimated 8,549 lbs. Since the GD Method is significantly different, I will use the average of the Torenbeek Method and the Class I estimate. The Class II fuselage weight estimate is 7,987 lbs.

14.2.1.4. Nacelle Weight

The nacelle weight consists of the podded engines and the structural weight associated with engine external ducts. The pylon weight is included. The nacelle weight also consists of the battery nacelles. The GD Method and the Torenbeek Method will be used to calculate the nacelle weight.

The GD Method for turbojet engines gives the nacelle weight as a function of the capture area per inlet in square feet, the number of inlets, the nacelle length, and the maximum static pressure at engine compressor face in psi (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[ W_n = 7.435 (N_{inl}) \left( A_{inl} \right)^{0.5} (l_n) (P_2)^{0.731} \]  

(95)

Using the hFan geometry, the fan diameter is 89 inches and the length is 156 inches (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011). I will assume that the maximum static pressure at the engine compressor face is 20 psi based on similar jet transport data. The GD Method estimates the nacelles to weigh 3,495 lbs.
The Torenbeek Method, which estimates the nacelle weight based on the turbofan engine thrust at takeoff estimates a nacelle weight of 1,564 lbs using the following equation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[ W_n = 0.065T_{TO} \]  

(96)

Taking into consideration the two estimation, along with the Class I estimation of 2,082 lbs, the Class II estimation of the nacelle weight is 2,381 lbs.

### 14.2.1.5. Landing Gear Weight

The landing gear weight for commercial transport aircraft is a function of the takeoff weight, for the GD Method calculation. For the Torenbeek Method, the landing gear is calculated considering constants in the landing gear weight equation. These constants are summarized in Table 39, below (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). The GD Method equations and Torenbeek Method equations are:

\[ W_g = 62.21 \left( \frac{W_{TO}}{1000} \right)^{0.84} \]  

(97)

\[ W_g = K_g (A_g + B_g (W_{TO})^3 + C_g W_{TO} + D_g (W_{TO})^{3/2} \]  

(98)

<table>
<thead>
<tr>
<th>Airplane Type</th>
<th>Gear Type</th>
<th>Gear Comp.</th>
<th>A_g</th>
<th>B_g</th>
<th>C_g</th>
<th>D_g</th>
</tr>
</thead>
<tbody>
<tr>
<td>Jet Trainers and Business Jets</td>
<td>Retr. Main</td>
<td>33.0</td>
<td>0.04</td>
<td>0.021</td>
<td>0.0</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Nose</td>
<td>12.0</td>
<td>0.06</td>
<td>0.0</td>
<td>0.0</td>
<td></td>
</tr>
<tr>
<td>Other civil airplanes</td>
<td>Fixed Main</td>
<td>20.0</td>
<td>0.10</td>
<td>0.019</td>
<td>0.0</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Nose</td>
<td>25.0</td>
<td>0.0</td>
<td>0.0024</td>
<td>0.0</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Tail</td>
<td>9.0</td>
<td>0.0</td>
<td>0.0024</td>
<td>0.0</td>
<td></td>
</tr>
<tr>
<td>Retr. Main</td>
<td>40.0</td>
<td>0.16</td>
<td>0.019</td>
<td>1.5x10^{-5}</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Nose</td>
<td>20.0</td>
<td>0.10</td>
<td>0.0</td>
<td>2.0x10^{-6}</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Tail</td>
<td>5.0</td>
<td>0.0</td>
<td>0.0031</td>
<td>0.0</td>
<td></td>
</tr>
</tbody>
</table>

The GD Method calculates the landing gear weight to be 2,406 lbs. Using the constants for retractable landing gear for a jet transport, the Torenbeek Method calculates the landing gear to
be 3,111 lbs. From the Class I estimate, landing gear was 2,654 lbs. The average results in 2,724 lbs for the Class II estimate.

14.2.1.6. Structure Weight Summary

The total structure weight is summarized by the following table:

Table 40. Class II Estimation of Structure Weight

<table>
<thead>
<tr>
<th>Component</th>
<th>GD Method Estimate (lbs)</th>
<th>Torenbeek Method Estimate (lbs)</th>
<th>Class I Estimate (lbs)</th>
<th>Class II Estimate (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>4,872</td>
<td>8,038</td>
<td>7,636</td>
<td>6,287</td>
</tr>
<tr>
<td>Empennage</td>
<td>3,628</td>
<td>2,263</td>
<td>1,192</td>
<td>2,627</td>
</tr>
<tr>
<td>Fuselage</td>
<td>4,024</td>
<td>7,424</td>
<td>8,549</td>
<td>7,987</td>
</tr>
<tr>
<td>Nacelle</td>
<td>3,495</td>
<td>1,564</td>
<td>2,082</td>
<td>2,381</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>2,406</td>
<td>3,111</td>
<td>2,654</td>
<td>2,724</td>
</tr>
<tr>
<td>TOTAL</td>
<td>18,425</td>
<td>22,400</td>
<td>22,113</td>
<td>22,006</td>
</tr>
</tbody>
</table>

14.2.2. Powerplant Weight

The powerplant weight, $W_{pwr}$, of the HTA consists of the following: weight of the engines, including the exhaust, cooling, supercharger, and lubrication systems; air induction system, which includes the inlet ducts, ramps, spikes, and controls; fuel system; and the propulsion system, which includes engine controls, starting systems, and provisions for engine installation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). Since the HTA will be using the hFan engine, which is a hybrid electric system, allowing the HTA to be able to use a gas turbine for some stages of the mission and batteries for other stages of the mission, while providing sufficient thrust for the aircraft with the added battery weight, the specifications of the propulsion system are available. A schematic of the hFan propulsion system is illustrated in the figure below (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011).
Figure 89. hFan engine overview

The HTA will require 2 suspended engines. Thus, from the manufacturers of the propulsion system, the weight of the propulsion system, which includes the advanced composite fan, 4-stage booster, ultra-high PR core compressor, combustor, thrust reverser, variable core nozzle, low and high pressure turbo, air induction system, fuel system, nacelles, and pylons, is 21,778 lbs (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011).

14.2.3. Fixed Equipment Weight

The fixed equipment weight, $W_{feq}$, consists of the weights of the flight control system, $W_{fc}$, hydraulic system, $W_{hs}$, the electric system, $W_{els}$, instrumentation, avionics, and electronics, $W_{iae}$, air-conditioning, pressurization, anti- and de-icing system, $W_{api}$, oxygen system, $W_{ox}$, auxiliary power unit (APU), $W_{apu}$, furnishings, $W_{fur}$, baggage and cargo handling equipment, $W_{bc}$, operational items, $W_{ops}$, and paint, $W_{pt}$ (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). Therefore, $W_{feq} = W_{fc} + W_{hs} + W_{els} + W_{iae} + W_{api} + W_{ox} + ...$
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\[ W_{apu} + W_{fur} + W_{bc} + W_{ops} + W_{pt}. \] The Torenbeek Method and GD Methods can be used to estimate the subsystem weights for the transport aircraft. The weights for each method will be compared to the weight estimated from similar data, and a Class II estimate will be formed.

14.2.3.1. Flight Control System Weight

The GD Method to estimate commercial transport aircraft flight control system weight involves the following equation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[
W_{fc} = 56.01 \left( \frac{W_{TO}(\bar{q})}{100000} \right)^{0.576} \tag{99}
\]

For the GD Method, the flight control system weight is a function of the design dive dynamic pressure in psf, \( \bar{q} \) and the takeoff weight. Using this method, 

\[
W_{fc} = 56.01 \left( \frac{77602.6(450)}{100000} \right)^{0.576} = 1633.4 \text{ lbs.}
\]

The Torenbeek Method estimates the flight control system weight for a jet transport with powered flight controls using the following equation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[
W_{fc} = 0.64(W_{TO})^{2/3} \tag{100}
\]

Thus, the Torenbeek Method estimate is \( W_{fc} = 1164.4 \text{ lbs.} \)

The SUGAR aircraft 765-093, 765-094, 765-095, and 765-096, which range from fully fuel-powered to hybrid powered aircraft, provide a database of additional weight data, including the flight control system data with which to compare the HTA flight control system weights (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011). The average flight control system weight is 2,932.5 lbs, which is about 1.8% of the takeoff weight. Using this ratio, the HTA is estimated to have a flight control system weight of 1,432 lbs.

Table 41. SUGAR Flight Control System Weight Data
Thus, combining the three methods of estimation, the Class II estimate of the flight control system weight is the average, 1,410 lbs.

**14.2.3.2. Hydraulic System Weight**

The hydraulic system weight can be estimated by using Roskam’s recommended range of $0.0060W_{TO} - 0.120W_{TO}$, giving a range of 466-931 lbs for the HTA (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). Additional comparisons can be performed, comparing the HTA to similar aircraft. When comparing to similar size aircraft, such as the 737-200, the Hydraulic system weight is 873 lbs, and using the ratio $\frac{W_{hs}}{W_{TO}} = 0.0076$, an estimate for the HTA’s $W_{hs}$ is 587 lbs.

When comparing to similarly powered aircraft, such as the SUGAR aircraft, the average for the SUGAR 765-093, 765-094, and 765-095 is 846 lbs (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011). For the SUGAR aircraft, the ratio of the hydraulic system weight to the takeoff weight is approximately 0.0053. Using this ratio and the takeoff weight of the HTA, an additional estimate for the HTA’s hydraulic system weight is 413 lbs.

<table>
<thead>
<tr>
<th>SUGAR A/C</th>
<th>A/C number</th>
<th>$W_{TO}$ (lbs)</th>
<th>$W_{fc}$ (lbs)</th>
<th>$\frac{W_{fc}}{W_{TO}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Free</td>
<td>765-093</td>
<td>175,635.000</td>
<td>3,084.000</td>
<td>0.018</td>
</tr>
<tr>
<td>Refined SUGAR</td>
<td>765-094</td>
<td>136,412.000</td>
<td>2,900.000</td>
<td>0.021</td>
</tr>
<tr>
<td>High</td>
<td>765-095</td>
<td>163,853.000</td>
<td>2,873.000</td>
<td>0.018</td>
</tr>
<tr>
<td>Volt</td>
<td>765-096</td>
<td>164,375.000</td>
<td>2,873.000</td>
<td>0.017</td>
</tr>
<tr>
<td>Average</td>
<td></td>
<td>2,932.500</td>
<td>0.018</td>
<td></td>
</tr>
</tbody>
</table>
Combining the estimates of 466 lbs, 931 lbs, 413 lbs, and 587 lbs, gives a Class II estimate of 600 lbs for the weight of the hydraulic system.

### 14.2.3.3. Electrical System Weight

The Torenbeek Method to estimate the electrical system weight uses an equation that is a function of the passenger cabin volume in cubic feet, $V_{pax}$. This equation is as follows (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

$$W_{els} = 10.8(V_{pax})^{0.7} \{1 - 0.018(V_{pax})^{0.35}\}$$  \hspace{1cm} (101)

The Torenbeek Method gives an estimate of 831 lbs.

The weight of the electrical system can be estimated from weights of similar electrical systems, as well (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011). The SUGAR aircraft electrical systems average 2362 lbs. Compared to their takeoff weight, the ratio of electrical system weight to takeoff weight is 0.015. Using this ratio, an estimate, from the SUGAR aircraft, for the electrical system weight of the HTA is 1152 lbs. Using the 737-200 data from Roskam, the ratio of electrical system weight to takeoff weight is 0.009, and the resulting weight for the HTA’s electrical system is 716 lbs.
Table 43. SUGAR Electrical System Weight Data

<table>
<thead>
<tr>
<th>SUGAR A/C</th>
<th>A/C number</th>
<th>$W_{TO}$ (lbs)</th>
<th>$W_{es}$ (lbs)</th>
<th>$W_{es}/W_{TO}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Free</td>
<td>765-093</td>
<td>175,635.000</td>
<td>2,557.000</td>
<td>0.015</td>
</tr>
<tr>
<td>Refined</td>
<td>765-094</td>
<td>136,412.000</td>
<td>2,297.000</td>
<td>0.017</td>
</tr>
<tr>
<td>SUGAR</td>
<td>765-095</td>
<td>163,853.000</td>
<td>2,297.000</td>
<td>0.014</td>
</tr>
<tr>
<td>Volt</td>
<td>765-096</td>
<td>164,375.000</td>
<td>2,297.000</td>
<td>0.014</td>
</tr>
<tr>
<td>Average</td>
<td></td>
<td>2,362.000</td>
<td></td>
<td>0.015</td>
</tr>
</tbody>
</table>

Averaging the resulting estimates from Torenbeek, SUGAR aircraft, and the 737-200, $W_{es} = 900$ lbs for the Class II estimate.

14.2.3.4. Instrumentation, Avionics, and Electronics Weight

The GD Method calculates the weight of the instruments as the sum of the flight instruments, engine instruments, and other instruments as a function of the number of engines and the number of pilots (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

$$W_i = N_{pit} \left( 15 + 0.032 \left( \frac{W_{TO}}{1000} \right) \right) + N_e \left( 5 + 0.006 \left( \frac{W_{TO}}{1000} \right) \right) + 0.15 \left( \frac{W_{TO}}{1000} \right) + 0.012W_{TO}$$ (102)

This method results in an estimate of 989 lbs.

The Torenbeek Method for jet transports is a function of the empty weight and the maximum range in nautical miles (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

$$W_{iae} = 0.575W_e^{0.556}R^{0.25}$$ (103)

The Torenbeek Method estimates $W_{iae}$ to be 1,475 lbs.
These two methods can now be compared to the data gathered from the SUGAR aircraft (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011). The SUGAR aircraft, have a ratio of $\frac{W_{iae}}{W_{TO}} = 0.014$, and therefore provide the HTA estimate for $W_{iae}$ to be 1,107 lbs. The Class II estimate for $W_{iae}$ is 1,050 lbs.

Table 44. SUGAR Data for Instrumentation, Avionics, and Electronics Weight

<table>
<thead>
<tr>
<th>SUGAR A/C</th>
<th>A/C number</th>
<th>$W_{TO}$ (lbs)</th>
<th>$W_{iae}$ (lbs)</th>
<th>$\frac{W_{iae}}{W_{TO}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Free</td>
<td>765-093</td>
<td>175,635.000</td>
<td>2,219.000</td>
<td>0.013</td>
</tr>
<tr>
<td>Refined SUGAR</td>
<td>765-094</td>
<td>136,412.000</td>
<td>2,277.000</td>
<td>0.017</td>
</tr>
<tr>
<td>High</td>
<td>765-095</td>
<td>163,853.000</td>
<td>2,277.000</td>
<td>0.014</td>
</tr>
<tr>
<td>Volt</td>
<td>765-096</td>
<td>164,375.000</td>
<td>2,277.000</td>
<td>0.014</td>
</tr>
<tr>
<td></td>
<td>Average</td>
<td>2,262.500</td>
<td></td>
<td>0.014</td>
</tr>
</tbody>
</table>

### 14.2.3.5. Air-Conditioning, Pressurization, and Anti- and Deicing Systems Weight

For pressurized commercial jet airplanes, the air-conditioning, pressurization system, and the anti- and deicing systems weight, $W_{api}$ can be estimated using the GD method. The GD Method is a function of the passenger cabin volume in cubic feet, $V_{pax}$, the number of crew members, $N_{cr}$, and the number of passengers, $N_{pax}$. The equation below provides the estimate (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

$$W_{api} = 469 \left( \frac{V_{pax}(N_{cr}+N_{pax})}{10,000} \right)^{0.419}$$ (104)

The resulting GD Method estimate for $W_{api}$ is 1,019 lbs.

The Torenbeek Method is a function of the length of the passenger cabin in feet, and is summarized by the following equation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).
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\[ W_{api} = 6.75(l_{pax})^{1.28} \]  

(105)

The Torenbeek Method estimate is 1,496 lbs.

Compared to similar aircraft, such as the 737-200, with \( W_{api} = 1,416 \) lbs, and SUGAR aircraft, which would give a ratio \( \frac{W_{api}}{W_{T0}} = 0.01 \), provides a low estimate of 792 lbs (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011). For the Class II estimate, the SUGAR aircraft data will be neglected, and \( W_{api} \) is estimated as the average of the GD and Torenbeek Methods along with the 737-200 data and SUGAR A/C data as 1,181 lbs.

Table 45. SUGAR API Weight Data

<table>
<thead>
<tr>
<th>SUGAR A/C</th>
<th>A/C number</th>
<th>( W_{T0} ) (lbs)</th>
<th>( W_{api} ) (lbs)</th>
<th>( \frac{W_{api}}{W_{T0}} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>SUGAR Free</td>
<td>765-093</td>
<td>175,635.000</td>
<td>1,796.000</td>
<td>0.010</td>
</tr>
<tr>
<td>Refined SUGAR</td>
<td>765-094</td>
<td>136,412.000</td>
<td>1,549.000</td>
<td>0.011</td>
</tr>
<tr>
<td>SUGAR High</td>
<td>765-095</td>
<td>163,853.000</td>
<td>1,582.000</td>
<td>0.010</td>
</tr>
<tr>
<td>SUGAR Volt</td>
<td>765-096</td>
<td>164,375.000</td>
<td>1,582.000</td>
<td>0.010</td>
</tr>
<tr>
<td>Average</td>
<td></td>
<td>1,627.250</td>
<td></td>
<td>0.010</td>
</tr>
</tbody>
</table>

14.2.3.6. Oxygen System Weight

The weight of the oxygen system can also be estimated using both the GD Method and Torenbeek Methods. The two equations, from Roskam are:

\[ W_{ox} = 7(N_{cr} + N_{pax})^{0.702} \]  

(106)

\[ W_{ox} = 30 + 1.2N_{pax} \]  

(107)

As a result, the GD Method gives an estimate of \( W_{ox} = 175 \) lbs, while the Torenbeek Method gives an estimate of \( W_{ox} = 145 \) lbs for short flights above 25,000 ft, and \( W_{ox} = \)
270 lbs for extended flights overwater. Since the HTA is a short to medium range transport aircraft, to determine a Class II estimate, $W_{ox} = 160$ lbs.

### 14.2.3.7. Auxiliary Power Unit Weight

The Auxiliary Power Unit (APU) weight will be taken into account, since the jet transport requires one. Although a specific APU has not been determined, the weight will be estimated using Roskam’s estimate of the takeoff weight, along with comparisons for the 737-200 and the SUGAR aircraft APU weights (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011).

According to Roskam, $W_{apu} = 0.004W_{TO} - 0.013W_{TO}$ (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). This range is 310-1008 lbs, with an average of 660 lbs. The SUGAR aircraft data gives an average APU weight of 1018.5 lbs, which is 0.6% of the takeoff weight. Using this data gives a low estimate of $W_{apu} = 498$ lbs. Taking the average of these values gives a Class II estimate of 450 lbs.

**Table 46. SUGAR APU Weight Data**

<table>
<thead>
<tr>
<th>SUGAR A/C</th>
<th>A/C number</th>
<th>$W_{TO}$ (lbs)</th>
<th>$W_{apu}$ (lbs)</th>
<th>$\frac{W_{apu}}{W_{TO}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>SUGAR Free</td>
<td>765-093</td>
<td>175,635</td>
<td>1,032</td>
<td>0.006</td>
</tr>
<tr>
<td>Refined SUGAR</td>
<td>765-094</td>
<td>136,412</td>
<td>1,014</td>
<td>0.007</td>
</tr>
<tr>
<td>SUGAR High</td>
<td>765-095</td>
<td>163,853</td>
<td>1,014</td>
<td>0.006</td>
</tr>
<tr>
<td>SUGAR Volt</td>
<td>765-096</td>
<td>164,375</td>
<td>1,014</td>
<td>0.006</td>
</tr>
<tr>
<td></td>
<td>Average</td>
<td>1,018.5</td>
<td></td>
<td>0.006</td>
</tr>
</tbody>
</table>

### 14.2.3.8. Furnishings Weight

Furnishings include cabin furniture and accessory items, like seats, insulation, trim panels, sound-proofing, instrument panels, control stands, lighting, and wiring. Furnishings also
include galley provisions and the structure, the lavatory system, overhead luggage containers, escape provisions, and fire-fighting equipment (Roskam, Airplane Design Part V: Component Weight Estimation, 2003). The Torenbeek Method and GD Method will provide initial estimates, with the SUGAR data and the 737-200 data providing additional support for the Class II Estimate.

The GD Method and Torenbeek Method provide the following equations, respectively to calculate $W_{fur}$.

$$ W_{fur} = 55N_{fdc} + 32N_{pax} + 15N_{cc} + k_{lav}(N_{pax})^{1.33} + k_{bag}(N_{pax})^{1.12} + 109\left(\frac{N_{pax}(1+P_{2})}{100}\right)^{0.505} + 0.771\left(\frac{W_{TO}}{1000}\right) $$

(108)

$$ W_{fur} = 0.211(W_{TO} - W_{p})^{0.91} $$

(109)

These two equations yield $W_{fur} = 3,918$ and $5,655$ lbs, respectively. The 737-200 aircraft gives an estimate of $W_{fur} = 4,463$ lbs, and using the ratio provided by the Sugar aircraft data, below, $W_{fur} = 4,651$ lbs.

Table 47. SUGAR Aircraft Furnishings Weight Data

<table>
<thead>
<tr>
<th>SUGAR A/C</th>
<th>A/C number</th>
<th>$W_{TO}$ (lbs)</th>
<th>$W_{fur}$ (lbs)</th>
<th>$\frac{W_{fur}}{W_{TO}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Free</td>
<td>765-093</td>
<td>175,635.000</td>
<td>10,866.000</td>
<td>0.062</td>
</tr>
<tr>
<td>Refined SUGAR</td>
<td>765-094</td>
<td>136,412.000</td>
<td>9,115.000</td>
<td>0.067</td>
</tr>
<tr>
<td>High</td>
<td>765-095</td>
<td>163,853.000</td>
<td>9,115.000</td>
<td>0.056</td>
</tr>
<tr>
<td>Volt</td>
<td>765-096</td>
<td>164,375.000</td>
<td>9,115.000</td>
<td>0.055</td>
</tr>
<tr>
<td></td>
<td>Average</td>
<td>9,552.750</td>
<td>0.060</td>
<td></td>
</tr>
</tbody>
</table>

Nikolai also provides a method to determine the weight of the furnishings. It is the sum of the crew seats, passenger seats, miscellaneous furniture, and the cabin windows. The equations that provide these, respectively are as follows (Nikolai, 1984).
Design of a Hybrid Transport Aircraft

\[
W_{cs} = 34.5(N_{cr})q^{0.25} \quad (110)
\]
\[
W_{ps} = 32.03(N_{pax}) \quad (111)
\]
\[
W_m = 0.771(W_{TO} \times 10^{-3}) \quad (112)
\]
\[
W_{cw} = 109.33(N_{pax}(1 + P_c) \times 10^{-2})^{0.505} \quad (113)
\]

\(W_{cs}, N_{cr}, q, W_{ps}, N_{pax}, W_m, W_{cw}, P_c\) are the weight of the crew’s seats, the number of crew, the dynamic pressure, weight of the passenger’s seats, number of passengers, weight of miscellaneous furnishings, weight of the cabin windows, and the ultimate cabin pressure in Psi, respectively. Using these equations, results in the following weights: \(W_{cs} = 318 \text{ lbs}, W_{ps} = 3,075 \text{ lbs}, W_m = 60 \text{ lbs}, \) and \(W_{cw} = 539 \text{ lbs},\) with a total of 3991 lbs.

The Class II weight estimate for furnishings weight will be 3,991 lbs.

14.2.3.9. Baggage and Cargo Handling Equipment Weight

According to Roskam, the baggage and cargo containers’ weight can be estimated using

\[
W_{bc} = 1.6 \times \text{lbs/ft}^3 \quad (114)
\]

There will be 3 containers at 80 ft\(^3\) each, for a total of 240 ft\(^3\), which gives the total weight to be 384 lbs, as the Class II estimate for \(W_{bc}\).

14.2.3.10. Operational Items Weight

The operational items include food, water, drinks, china, and lavatory supplies. Using the SUGAR aircraft database, below, the Class II estimate of the HTA’s operational items weight will be calculated (Bradley & Droney, Subsonic Ultra Green Aircraft Research: Phase I Final Report, 2011). Assuming the operational items weight as 4.6% the takeoff weight, the HTA’s estimate for operational items weight from SUGAR A/C data is \(W_{ops} = 3,540 \text{ lbs}\).
Table 48. SUGAR Data for Operational Items Weight

<table>
<thead>
<tr>
<th>SUGAR A/C</th>
<th>A/C number</th>
<th>$W_{TO}$ (lbs)</th>
<th>$W_{ops}$ (lbs)</th>
<th>$\frac{W_{ops}}{W_{TO}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Free</td>
<td>765-093</td>
<td>175,635.000</td>
<td>7,342.000</td>
<td>0.042</td>
</tr>
<tr>
<td>Refined SUGAR</td>
<td>765-094</td>
<td>136,412.000</td>
<td>7,207.000</td>
<td>0.053</td>
</tr>
<tr>
<td>High</td>
<td>765-095</td>
<td>163,853.000</td>
<td>7,207.000</td>
<td>0.044</td>
</tr>
<tr>
<td>Volt</td>
<td>765-096</td>
<td>164,375.000</td>
<td>7,207.000</td>
<td>0.044</td>
</tr>
<tr>
<td></td>
<td>Average</td>
<td>7,240.750</td>
<td>0.046</td>
<td></td>
</tr>
</tbody>
</table>

According to Nikolai’s Fundamental’s of Aircraft Design,

$$W_{lav} = 1.11 * N_{pax}^{1.33}$$  \hspace{1cm} (115)

$$W_{food} = K_{BUF} * N_{pax}^{1.12}$$  \hspace{1cm} (116)

where $W_{lav}$ is the weight of the lavatories and water provisions, $W_{food}$ is the weight of the food provisions, and $K_{BUF} = 5.68$ or $1.02$ for long range flights and short range flights, respectively (Nikolai, 1984). Combining these weights for 96 passengers give an estimate for the operational weight of 650 lbs. This is significantly less than the data from the SUGAR A/C. Differences include the number of passengers, and the length of the flight.

The calculated value of 650 lbs will be used as the estimate for the Class II operational weight.

14.2.3.11. Weight of Paint

Roskam provides the following equation to estimate the weight of paint for a well painted airplane to be:

$$W_{pt} = 0.003W_{TO} - 0.006W_{TO}$$  \hspace{1cm} (117)
This gives a range of 233-466 lbs and an average of 349 lbs. The 737-700 uses approximately 350 lbs of paint (Hansen, n.d.). The Class II estimate will use an equivalent ratio of paint equal to 260 lbs.

### 14.2.3.12. Fixed Equipment Weight Summary

**Table 49. Fixed Equipment Weight Summary**

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Flight Control System</td>
<td>1633.4</td>
<td>1164.4</td>
<td>1432</td>
<td></td>
<td></td>
<td></td>
<td>1,410</td>
</tr>
<tr>
<td>Hydraulic System</td>
<td></td>
<td>466-931</td>
<td>873</td>
<td>413</td>
<td></td>
<td></td>
<td>600</td>
</tr>
<tr>
<td>Electrical System</td>
<td></td>
<td>831</td>
<td>716</td>
<td>1,152</td>
<td></td>
<td></td>
<td>900</td>
</tr>
<tr>
<td>Instrumentation, Avionics, and Electronics</td>
<td>989</td>
<td>1,475</td>
<td>1,107</td>
<td></td>
<td></td>
<td></td>
<td>1,050</td>
</tr>
<tr>
<td>Air-Conditioning, Pressurization, Anti and De-icing</td>
<td>1,019</td>
<td>1,496</td>
<td>1,416</td>
<td>793</td>
<td></td>
<td></td>
<td>1,181</td>
</tr>
<tr>
<td>Oxygen System</td>
<td>175</td>
<td>145-208</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>160</td>
</tr>
<tr>
<td>APU</td>
<td></td>
<td>310-1008</td>
<td>498</td>
<td></td>
<td>450</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Furnishings</td>
<td>3,919</td>
<td>5,655</td>
<td>4,463</td>
<td>4,652</td>
<td>3,991</td>
<td>3,991</td>
<td></td>
</tr>
<tr>
<td>Baggage and Cargo</td>
<td></td>
<td>384</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>384</td>
</tr>
<tr>
<td>Operational Items</td>
<td></td>
<td></td>
<td>3,540</td>
<td>650</td>
<td>650</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Paint</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>260</td>
</tr>
<tr>
<td>TOTAL</td>
<td>233-466</td>
<td>350</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>11,036</td>
</tr>
</tbody>
</table>

### 14.2.4. Empty Weight Discussion

The empty weight is the sum of the structural weight, powerplant weight, and fixed equipment weight. \( W_e = W_{struct} + W_{pwr} + W_{feq} \). Combining the totals from each of these
sections gives a total estimate of 52,439 lbs for the empty weight. Note that the nacelle weights were accounted for in the powerplant weight calculations, and the structural nacelle weight estimate will be neglected, since the manufacturer’s data is available for the powerplant.

For the HTA, which uses fuel only in stages 1-4, and is battery powered for the remaining stages, the Class I estimate for the empty weight is 48,379 lbs. This difference of 4,060 lbs is fairly close to the Class I estimate, and gives a 7.7% difference.

The Class II weight sizing will need to be iterated if the weight savings criteria that follows does not provide a convergence. The weight savings criteria involve manufacturing the fuselage from composites, which would save 15% of the weight, manufacturing parts of the wing and fuselage under low stress from composites, which would save about 5%, and using a quadruplex digital flight control system and fly-by-wire as opposed to mechanical flight controls, which would save 15%, and replacing the primary wing and fuselage structure with lithium aluminum, which would give a 6% weight savings (Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System, 2011). Thus, the empennage weight could be reduced by 394 lbs; the wing and fuselage low-stress composite replacement could be reduced by 713 lbs; the flight controls system could be reduced by 212 lbs; and the lithium aluminum would reduce the wing and fuselage structure by 857 lbs. Thus, the estimated empty weight has been lowered to 50,263 lbs, now only a 3.7% difference.

Considering the percent difference is small, and the Class I weight sizing will continue to be used.

14.3. Component Centers of Gravity

The c.g. locations for the structural components, powerplant, and fixed equipment components will be presented using Roskam’s guidelines. The table below illustrates the c.g.
location of the structural components, powerplant, and fixed equipment (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

Using the revised Class II weights and the c.g. locations for both the x and z direction gives the following weights summary.

*Table 50. Class II Weights and CG locations*

<table>
<thead>
<tr>
<th>No.</th>
<th>Type of Component</th>
<th>Wi (lbs)</th>
<th>xi (in.)</th>
<th>Wixi (in lbs)</th>
<th>yi (in.)</th>
<th>Wiyi (in lbs)</th>
<th>zi (in.)</th>
<th>Wizi (in lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Fuselage Group</td>
<td>7,130</td>
<td>641</td>
<td>4,570,330</td>
<td>0</td>
<td>0</td>
<td>248</td>
<td>1,768,240</td>
</tr>
<tr>
<td>2</td>
<td>Wing Group</td>
<td>5,568</td>
<td>680</td>
<td>3,786,240</td>
<td>0</td>
<td>0</td>
<td>350</td>
<td>1,948,800</td>
</tr>
<tr>
<td>3</td>
<td>Empennage Group</td>
<td>2,233</td>
<td>1190</td>
<td>2,657,270</td>
<td>0</td>
<td>0</td>
<td>300</td>
<td>669,900</td>
</tr>
<tr>
<td>4</td>
<td>Engine Group</td>
<td>21,778</td>
<td>590</td>
<td>12,849,020</td>
<td>0</td>
<td>0</td>
<td>140</td>
<td>3,048,920</td>
</tr>
<tr>
<td>5a</td>
<td>Landing Gear Group: Nose Gear</td>
<td>500</td>
<td>390</td>
<td>195,000</td>
<td>0</td>
<td>0</td>
<td>130</td>
<td>65,000</td>
</tr>
<tr>
<td>5b</td>
<td>Landing Gear Group: Main Gear</td>
<td>2224</td>
<td>710</td>
<td>1,579,040</td>
<td>0</td>
<td>0</td>
<td>130</td>
<td>289,120</td>
</tr>
<tr>
<td>6</td>
<td>Fixed Equipment Group</td>
<td>10,824</td>
<td>680</td>
<td>7,360,320</td>
<td>0</td>
<td>0</td>
<td>200</td>
<td>2,164,800</td>
</tr>
<tr>
<td>7</td>
<td>Trapped Fuel and Oil</td>
<td>34.4</td>
<td>690</td>
<td>23,736</td>
<td>0</td>
<td>0</td>
<td>215</td>
<td>7,396</td>
</tr>
<tr>
<td>8</td>
<td>Crew</td>
<td>700</td>
<td>300</td>
<td>210,000</td>
<td>0</td>
<td>0</td>
<td>240</td>
<td>168,000</td>
</tr>
<tr>
<td>9</td>
<td>Fuel</td>
<td>3,440</td>
<td>690</td>
<td>2,373,600</td>
<td>0</td>
<td>0</td>
<td>430</td>
<td>1,479,200</td>
</tr>
<tr>
<td>10</td>
<td>Batteries</td>
<td>12,000</td>
<td>733</td>
<td>8,796,000</td>
<td>0</td>
<td>0</td>
<td>120</td>
<td>1,440,000</td>
</tr>
<tr>
<td>11</td>
<td>Passengers</td>
<td>16,800</td>
<td>680</td>
<td>11,424,000</td>
<td>0</td>
<td>0</td>
<td>200</td>
<td>3,360,000</td>
</tr>
<tr>
<td>12</td>
<td>Luggage</td>
<td>3,000</td>
<td>680</td>
<td>2,040,000</td>
<td>0</td>
<td>0</td>
<td>200</td>
<td>600,000</td>
</tr>
</tbody>
</table>

The x c.g. location is located at 55.9 ft, and the z c.g. location is located at 16.4 ft based on the equations from Roskam (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).
If any of the components are moved, then the c.g. location moves according to the following equation (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[
\Delta x_{cg} = \frac{\Delta x_i(W_i)}{\sum_{i=1}^{n} W_i}
\]

The most likely movement could be the batteries, but since they are suspended below the wings, they do not have a wide range of movement available.

The c.g. excursion plot, based on the Class II data, is illustrated in the figure below. The plot illustrates that the most forward c.g. location is at 57.33 ft and the most aft c.g. location is at 72.11 ft. This was plotted by varying the number of passengers, luggage, and fuel. The conditions included a fully loaded aircraft, an aircraft with half passenger and full fuel, all passengers and half the luggage, zero passengers and full luggage, no passengers and no cargo, zero fuel, passengers only, and cargo only.

![c.g. Excursion Diagram](image)

*Figure 90. Class II cg excursion diagram*
14.4. Airplane Inertias

To estimate the moments and products of inertia for the HTA, the weights from the Class II sizing and the c.g. locations in feet will be used. The engine group will be one weight and c.g. location, and will therefore be a ‘lumped mass’ assumption. This is also true for the passengers and fuel systems. The structural components will also be assumed to have uniform mass distributions. The following table summarizes the moments and products of inertia.

Table 51. Moments and products of Inertia

<table>
<thead>
<tr>
<th>$I_{xx}$</th>
<th>$I_{yy}$</th>
<th>$I_{zz}$</th>
<th>$I_{xy}$</th>
<th>$I_{yz}$</th>
<th>$I_{zx}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>111,166.12</td>
<td>314,123.14</td>
<td>202,957.02</td>
<td>0</td>
<td>0</td>
<td>-52,406.14</td>
</tr>
</tbody>
</table>

The table was computed using the following equations (Roskam, Airplane Design Part V: Component Weight Estimation, 2003).

\[
I_{xx} = \sum_{i=1}^{n} m_i \left( (y_i - y_{cg})^2 + (z_i - z_{cg})^2 \right) \tag{121}
\]

\[
I_{yy} = \sum_{i=1}^{n} m_i \left( (z_i - z_{cg})^2 + (x_i - x_{cg})^2 \right) \tag{122}
\]

\[
I_{zz} = \sum_{i=1}^{n} m_i \left( (x_i - x_{cg})^2 + (y_i - y_{cg})^2 \right) \tag{123}
\]

\[
I_{xy} = \sum_{i=1}^{n} m_i \left( (x_i - x_{cg})(y_i - y_{cg}) \right) \tag{124}
\]

\[
I_{xz} = \sum_{i=1}^{n} m_i \left( (y_i - y)(z_i - z_{cg}) \right) \tag{125}
\]

\[
I_{zx} = \sum_{i=1}^{n} m_i \left( (z_i - z_{cg})(x_i - x_{cg}) \right) \tag{126}
\]

15. Cost Analysis

The life cycle cost consists of the research and development costs, production and construction costs, operations and maintenance costs, and retirement and disposal costs. For typical aircraft, the operating costs compose nearly 94% of the life cycle cost (Ploetner,
Schmidt, Baranowski, Isikveren, & Hornung, 2013). Thus, the preliminary cost analysis for the HTA will consider the operating cost (Roskam, Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System, 2011).

In order to estimate the program operating cost, $C_{ops}$, direct operating cost, DOC, and indirect operating cost, IOC, mission data will be reviewed. These calculations will require the block distance, $R_{bl}$, block time, $T_{bl}$, block speed, $V_{bl}$, and annual utilization in block hours, $U_{ann_{bl}}$. The block distance can be considered the door to door distance. This distance will be the 1,720 nm established for the HTA. The block time is the door to door time from the departure gate to the parking spot. The block time is calculated through the following equations (Roskam, Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing, and Operating, 2015).

$$T_{bl} = T_{gm} + T_{ct} + T_{cr} + T_{de} \quad (127)$$

$$T_{gm} = 0.51(10)^{-6}(W_{TO}) + 0.125 \quad (128)$$

$$T_{cr} = (1.06R_{bl} - R_{ct} - R_{de} + R_{man})/V_{cr} \quad (129)$$

$$R_{ct} = (V_{ct})(T_{ct}) \quad (130)$$

$$R_{man} = (V_{man})(T_{man}) \quad (131)$$

$$T_{man} = 0.25(10^{-6})W_{TO} + 0.0625 \quad (132)$$

The times consisting of the time spent on ground maneuvers, $T_{gm}$, such as leaving the gate, taxiing to the runway, time on the takeoff run, on the landing ground run, and taxiing to the destination, time spent in climb, $T_{ct}$, time spent in cruise, $T_{cr}$, and time spent in descent, $T_{de}$. The ground maneuver time depends on the takeoff weight, and the time for cruise depends on the block distance, $R_{bl}$, distance for climb, $R_{ct}$, distance for descent, $R_{de}$, maneuvering distance, $R_{man}$, and the cruise velocity, $V_{cr}$. The distance for climb depends on the climb velocity and the
time to climb. The distance for maneuvering depends on the maneuvering velocity, which will be 250 kts, and time spent doing the maneuver. The results of the calculations are as follows:

**Table 52. Block Time Results**

<table>
<thead>
<tr>
<th>Time segment</th>
<th>$T_{bl}$</th>
<th>$T_{gm}$</th>
<th>$T_{cr}$</th>
<th>$T_{de}$</th>
<th>$T_{el}$</th>
<th>$T_{man}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Time (hrs)</td>
<td>4.485</td>
<td>0.165</td>
<td>3.554</td>
<td>0.267</td>
<td>0.5</td>
<td>0.082</td>
</tr>
</tbody>
</table>

**Table 53. Block Speed Results**

<table>
<thead>
<tr>
<th>Speed (kts)</th>
<th>$V_{bl}$</th>
<th>$V_{de}$</th>
<th>$V_{man}$</th>
<th>$V_{cr}$</th>
<th>$V_{de}$</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>383.478</td>
<td>250</td>
<td>250</td>
<td>500</td>
<td>250</td>
</tr>
</tbody>
</table>

**Table 54. Block Distance Results**

<table>
<thead>
<tr>
<th>Distance (nm)</th>
<th>$R_{bl}$</th>
<th>$R_{de}$</th>
<th>$R_{man}$</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>1720</td>
<td>66.667</td>
<td>20.475</td>
</tr>
</tbody>
</table>

Thus, the annual utilization in block hours can be calculated using the following equation (Roskam, Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing, and Operating, 2015).

$$U_{ann,bl} = 10^3 \left[3.4546 \cdot T_{bl} + 2.994 \right. - \left. (12.289(T_{bl})^2 - 5.6626(T_{bl}) + 8.964)^{\frac{1}{2}} \right] (133)$$

The annual utilization is approximately 3,297 hours, which is equivalent to about 9 hours of daily utilization. These results are comparable to other aircraft, including 727-200, 737, 757, and DC-8. For these medium range aircraft, with $2 < T_{bl} < 5$, the annual utilization numbers in block hours ranges from 2,100-3,300 block hours. The HTA falls in this range. Now that these parameters have been calculated, the direct operating cost can be calculated.

### 15.1. Direct Operating Cost

The direct operating cost is the sum of the direct operating cost of flying, maintenance, depreciation, landing fees, and financing, abbreviated as $DOC, DOC_{flt}, DOC_{maint}, DOC_{depr}$. 
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$DOC_{inr}$, and $DOC_{fin}$, respectively. Each of these direct operating costs will be measured in USD/nm. Each of these will be calculated and summed to find the total direct operating cost.

15.1.1. Direct Operating Cost of Flying

The direct operating cost of flying consists of the sum of the cost of the crew, $C_{crew}$, cost of the fuel and oil, cost of the batteries, and cost of the airframe insurance.

15.1.1.1. Cost of the Crew

The cost of the crew depends on the number of crew members, $n_{c_j}$, the type of crew member, with $n_{c_1}$ being the captain and $n_{c_2}$ being the co-pilot, the factor, $k_j$, which accounts for vacation pay, cost of training, crew insurance and premiums, and payroll tax, the block speed, $V_{bl}$, the annual salaries of the crew members, $SAL_j$, the number of flight hours per year for a crew member, $AH_j$, and the travel expense factor, $TEF_j$.

$$C_{crew} = \sum_{j=1}^{j} \left[ n_{c_j} \left( \frac{1+k_j}{V_{bl}} \right) \left( \frac{SAL_j}{AH_j} \right) + \left( \frac{TEF_j}{V_{bl}} \right) \right]$$

All the of the costs are for 2018. They have been scaled using the cost escalation factor, CEF, which is

$$CEF_{year} = 6.008930 + 0.10280(year - 2014)$$

There will be two crew members for the HTA, a captain and co-pilot. The salaries of each for the respective crew members will be assumed to be $158,686 and $102,679. Flight crew will be assumed to fly an average of 800 flight hours per year. The travel expense factor for each of the crew is 13.06 per block hour. Thus, the total cost of the crew is 1.14 USD/nm.

15.1.1.2. Cost of the Fuel and Oil

The cost of fuel and oil, $C_{foil}$, depends on the mission fuel weight, the price of the fuel, FP, and the density of the fuel, FD.
The current fuel price is $2.20 per gallon (Jet Fuel Daily Price, 2018). The fuel density is 6.74 lbs/gallon (Roskam, Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing, and Operating, 2015). Using a mass fuel fraction of 0.955696, the weight of the fuel, and thus the weight of the block fuel is 3,438 lbs. The total cost of fuel and oil in USD per nautical mile is 0.69 USD/nm.

15.1.1.3. Cost of the Batteries

To determine the direct operating cost for the batteries, $DOC_{bat}$, measured in USD/nm, the following assumptions will be made. The flight will require 35,314 kWh for the flight. Generally, the cost per charge per kWh is $0.0085. Thus the rate in USD/nm for the batteries charge is 0.17 USD/nm. Note that the initial cost of the battery is not included in this.

15.1.1.4. Summary of Direct Operating Cost of Flying

The direct operating cost of flying, $DOC_{flt}$, can be calculated using the following equation (Roskam, Airplane Design Part VII: Determination of Stability, Control and Performance Characteristics: FAR and Military Requirements, 2006).

$$DOC_{flt} = C_{crew} + C_{pol} + C_{ins} + C_{bat}$$

Since the cost of insurance is 2% of the direct operating cost, the direct operating cost of flying if a function of the direct operating cost. $DOC_{flt} = 2.10 + 0.02(DOC)$ USD/nm.

15.1.2. Direct Operating Cost of Maintenance

The direct operating cost of maintenance is calculated using the following equation, from Roskam. The direct operating cost of maintenance is the sum of the labor cost of the airframe and systems maintenance, labor cost of the engine maintenance, cost of maintenance materials for the airframe and system, cost of maintenance materials for the engines, and applied
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maintenance burden, all measured in USD/nm. The following equations summarize the
calculations necessary to determine the direct operating cost of maintenance (Roskam, Airplane
Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing, and
Operating, 2015).

\[
DOC_{\text{maint}} = C_{\text{lab/ap}} + C_{\text{lab/eng}} + C_{\text{mat/ap}} + C_{\text{mat/eng}} + C_{\text{amb}} \tag{138}
\]

\[
C_{\text{lab/ap}} = 1.03(MHR_{\text{mapbl}})(R_{i_{\text{ap}}})/(V_{bl}) \tag{139}
\]

\[
C_{\text{lab/eng}} = 1.03(1.3)N_{e}(MHR_{\text{menGbbl}})(R_{i_{\text{eng}}})/(V_{bl}) \tag{140}
\]

\[
C_{\text{mat/ap}} = 1.03(C_{\text{mat/apblhr}})V_{bl} \tag{141}
\]

\[
C_{\text{mat/eng}} = 1.03 * 1.3 * N_{e} * C_{\text{mat/engblhr}}/V_{bl} \tag{142}
\]

\[
C_{\text{amb}} = 1.03 \{ (f_{\text{amb/lab}}) [(MHR_{\text{mapbl}})(R_{i_{\text{ap}}}) + (N_{e})(MHR_{\text{menGbbl}})(R_{i_{\text{eng}}})] +
(f_{\text{amb/lab}})[C_{\text{mat/apblhr}} + (N_{e})C_{\text{mat/engblhr}}] /V_{bl} \} \tag{143}
\]

The maintenance labor cost for the airframe and system depends on the number of
maintenance manhours per blockhour for airframe systems, which is 6.35 hours, the weight of
the airframe, which is 26,601 lbs, and the airplane maintenance labor rate per manhour, of 16
USD/nm. These inputs produce \( C_{\text{lab/ap}} = 0.3142 \) USD/nm.

The cost of the maintenance labor for the engines depends on the number of engines, the
number of engine maintenance hours per engine, which will be assumed to be 5 hours, the hourly
labor rate for engine maintenance, at $18.40, and the block speed of 383.5 kts. This produces
\( C_{\text{lab/eng}} = 0.64 \) USD/nm.

The cost of maintenance materials for the airframe and systems, other than the engines,
depends on the airframe price, and the cost of the maintenance materials per blockhour, at 100
USD/nm. Thus, \( C_{\text{mat/ap}} = 0.27 \) USD/nm.
The cost of maintenance materials for the engines depends on the number of engines, the cost of the maintenance materials per blockhour, which is 90.34 USD/blockhour, and the block speed of 383.5 kts. \( C_{\text{mat/eng}} = 0.63 \text{ USD/nm} \).

Finally, the cost of the applied maintenance burden per nm is a function of overhead distribution factors for labor and material cost, the time of flight, the block time, and the cost of the maintenance man hours per flight for the materials. This amounts to 1.053 USD/nm.

Taking the sum of these costs gives the direct operating cost of maintenance to be 2.91 USD/nm.

15.1.3. Direct Operating Cost of Depreciation

The direct operating cost of depreciation is composed of the cost of the airplane depreciation without engines, avionics systems, the cost of the engine depreciation, the cost of the avionics system depreciation, the cost of the depreciation of the spare parts, and the cost of the depreciation of the engine spare parts (Roskam, Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing, and Operating, 2015).

\[
DOC_{\text{depr}} = C_{\text{dap}} + C_{\text{deng}} + C_{\text{dav}} + C_{\text{dasp}} + C_{\text{dengsp}}
\] (144)

The airframe depreciation per nautical mile is calculated using the following equation.

\[
C_{\text{dap}} = \frac{\left[ F_{\text{dap}} \right] \left[ (\text{AEP}) - (N_{\text{a}})(\text{EP}) - (N_{\text{p}})(\text{PP}) - (\text{ASP}) \right]}{(\text{DP}_{\text{ap}})(U_{\text{annh}})(V_{\text{bl}})}
\] (145)

This cost depends on the airframe depreciation factor, which is 0.85, the airplane estimated price of $33,354,004 for 2018, the number of engines, 2, the engine price, which will be assumed to be $1,558,607, since the price of the hFan engine is not available, but this price is comparable to other engines, the avionics system price of $2,670,000, the depreciation period off the airplane of 10 years, the annual utilization in block hours, calculated as 3,297 hours, and the block speed, 383.5 kts. This gives \( C_{\text{dap}} = 1.85 \text{ USD/nm} \).
The cost of engine depreciation per nautical mile is calculated using:

\[
C_{\text{deng}} = \frac{(F_{\text{deng}})_{N_e}(EP)}{(DP_{\text{eng}})(U_{\text{ann}})(V_{\text{bl}})}
\]  \hspace{1cm} (146)

This equation depends on the engine depreciation factor of 0.85, the engine depreciation period of 7 years, which may change when using the new hybrid engines, and other previously established factors of engine price, annual utilization in block hours, and block speed. The result is that \( C_{\text{deng}} = 0.299 \) USD/nm.

The cost of the depreciation of the avionics system in USD per nautical mile is calculated using:

\[
C_{\text{dav}} = \frac{(F_{\text{dav}})_{\text{ASP}}}{(DP_{\text{av}})(U_{\text{ann}})(V_{\text{bl}})}
\]  \hspace{1cm} (147)

Thus, \( C_{\text{dav}} = 0.42 \) USD/nm with an avionics system depreciation factor of 1 and the depreciation time to be 5 years.

The cost of the depreciation of airplane spare parts in USD/nm is calculated using:

\[
C_{\text{dasp}} = \frac{(F_{\text{dasp}})(F_{\text{asp}})(AEP-(N_e)(EP))}{(DP_{\text{asp}})(U_{\text{ann}})(V_{\text{bl}})}
\]  \hspace{1cm} (148)

Thus \( C_{\text{dasp}} \) is a function of the airplane spare parts depreciation factor of 0.85, the airplane spare parts factor of 1, the airplane spare parts depreciation period of 10 years, and other previously calculated values. \( C_{\text{dasp}} = 0.20 \) USD/nm.

The cost of the depreciation of engine spare parts is given by:

\[
C_{\text{dasp}} = \frac{(F_{\text{deng}})(F_{\text{eng}})(N_e)(EP)(ESPFF)}{(DP_{\text{eng}})(U_{\text{ann}})(V_{\text{bl}})}
\]  \hspace{1cm} (149)

\( C_{\text{dasp}} = 0.23 \) USD/nm, and is based on the engine spare parts depreciation factor of 0.85, the engine spare parts factor of 0.5, the engine spare parts price factor of 1.50, the depreciation period for the engine spare parts, which is 7 years, and the block speed and annual utilization.
Summarizing, \( \text{DOC}_{\text{depr}} = C_{\text{dap}} + C_{\text{deng}} + C_{\text{dav}} + C_{\text{dasp}} + C_{\text{dengsp}} = 1.85 + 0.299 + 0.42 + 0.20 + 0.23 = 3.01 \text{ USD/nm} \).

15.1.4. Direct Operating Cost of Landing Fees, Navigation Fees, and Registry Taxes

The direct operating cost of landing fees, navigation fees, and registry taxes is given by the following equations (Roskam, Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing, and Operating, 2015).

\[
\begin{align*}
\text{DOC}_{\text{lmr}} &= C_{\text{lf}} + C_{\text{nF}} + C_{\text{rt}} \quad (150) \\
C_{\text{lf}} &= \left( C_{\text{apf}} \right) / (V_{\text{bi}} T_{\text{bi}}) \quad (151) \\
C_{\text{nF}} &= \left( C_{\text{apnF}} \right) / (V_{\text{bi}} T_{\text{bi}}) \quad (152) \\
C_{\text{rt}} &= f_{\text{rt}} (\text{DOC}) \quad (153)
\end{align*}
\]

The landing fee cost is 0.09 USD/nm, with the airplane landing fee per landing being 155.21 USD/nm. The cost of navigation fee will be assumed to be $0 USD/nm as the plane will not fly internationally. The cost of financing will be assumed to be 0.58 USD/nm, and the cost of registry taxes is 0.01 USD/nm. Thus, the resulting sum is 0.099 USD/nm.

15.1.5. Direct Operating Cost of Financing

The direct operating cost of financing, \( \text{DOC}_{\text{fin}} \) is 7% of the DOC. Thus, this amounts to approximately 0.34 USD/nm.

15.1.5. Direct Operating Cost Discussion

The direct operating cost, totals 8.45 USD/nm. The following chart illustrates the breakdown of the DOC for a jet transport aircraft.
Based off of the calculations for the direct operating cost for the HTA, the figures below illustrate the breakdown of the DOC for both the HTA and a fuel version of the HTA. There is a significant decrease in fuel usage, but higher maintenance due to the batteries. Additionally, there is an increase in the depreciation with batteries due to the life cycles.

Figure 91. DOC for a Jet Transport

Figure 92. HTA DOC with batteries and fuel
Additionally, the figure below illustrates the decrease in the direct operating cost over the years of the flight (Maddalon, Molloy, & Neubauer, Jr., 1980). This figure is for a jet transport. For a hybrid aircraft, the DOC should decrease more quickly as there will be less fuel to purchase. Note that this chart extends for 15 years. Many of these calculations were performed assuming a 10 year life of the airplane. However, many aircraft last longer than 10 years, which only increases the profit over the years.

Sensitivity studies on a fully electric jet transport demonstrate that fuel price and electricity price have the strongest impact of the DOC. The study shows that with a 10%
increase in fuel costs, there is a 3% increase in DOC. Additionally, a reduction in the number of cycles of battery use, less than the typical 2000 cycles, combined with battery installation costs also produces higher DOCs (Ploetner, Schmidt, Baranowski, Isikveren, & Hornung, 2013).

15.2. Indirect Operating Cost

The indirect operating cost, $IOC$, is composed of the indirect operating cost for passenger services, maintaining depreciating ground equipment and facilities, airplane and traffic servicing, promotions sales, and entertainment, and general administrative expenses. Its units are in USD/nm. The costs for these services can be estimated through the equation:

$$IOC = F_{ioc}(DOC)$$

$(154)$

$F_{ioc}$ is the factor extrapolated from the 737 data from Roskam. This factor is 0.5. Thus, the indirect operating cost is 4.14 USD/nm.

15.3. Program Operating Cost

The program operating cost depends on the indirect program operating costs, the direct program operating costs, and the number of aircraft acquired. Due to the fact that operating costs vary between customers, the program operating costs depends on the number of customers. For the purposes of these calculations, 1,200 aircraft will be assumed to be in acquisition (Sales, 2018). As of January 1, 2018, Boeing has orders for more than 4,300 737- MAX aircraft. Since the HTA has similar mission specifications, a high volume will be assumed for acquisition. Then the program operating cost can be estimated using the following equations (Roskam, Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing, and Operating, 2015).

\[
C_{ops} = \sum_{i=1}^{n} \left[ (C_{ops_{dir}})_i (N_{acq})_i + (C_{ops_{ind}})_i (N_{acq})_i \right]
\]

$(155)$

\[
(C_{ops_{dir}})_i = (DOC)_i (R_{blann})_i (N_{yr})_i
\]

$(156)$
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\[
(c_{ops,ind})_i = (IOC)_i (R_{biann})_i (N_{yr})_i
\] (157)

Assuming the aircraft are in use for 10 years, the operating cost is $190,835,543,254.38. The early design has a significant impact on the ultimate cost of the aircraft. In the future, trade studies can be performed on the number of years in service and the number of aircraft acquired per year to determine the effect on the operating cost.

### 15.3. Hybrid Power Cost Analysis

Although the price of jet fuel fluctuates per gallon, the general trend illustrates an increase in fuel costs and an increase in the number of general aviation aircraft, as discussed in the introduction. The price of electricity to recharge the batteries is significantly cheaper than jet fuel and has the added benefit of drastically reducing carbon dioxide and other harmful emittants. Electricity costs approximately 8.5 cents per kilowatt hour. For a fully-fuel powered version of the HTA, which requires 17,888 lbs or 2,670 gallons of jet fuel, this would reduce the cost of the total fuel per flight from $5,874 to $1,129.

#### 15.3.1. Sample Battery Specifications

Although battery technology is not yet at its peak in power, companies are selling advanced lithium ion batteries for aviation power. True Blue Power is one such company.

For aviation, True Blue has a 46 amp hour battery, which at an output voltage of 26.4 Volts, and weight of approximately 23.4 kg, gives approximately 51.7 Wh/kg of energy. The low specific energy density would result in more batteries and higher weight, further illustrating the need for increased battery energy density. However, with the promise of better battery technology and the capabilities of lithium ion batteries, these batteries may provide future capabilities.
This battery has a life expectancy of 8 years with a 2 year maintenance interval. It weighs 51.7 lbs and has superior performance at extreme temperatures, including -40ºF to 158ºF and at pressures up to 35 Psi. Its dimensions are 10.9 in by 10.5 in by 10.1 in, making it easily fit in small spaces and configure with other batteries (True Blue Power, 2018). Each battery costs approximately $16,400 (True Blue Power, 2018). Though the batteries may initially be expensive, this one time cost for the life of the aircraft will drastically decrease the cost to power the aircraft over time.

16. Conclusion

The preliminary design of the narrow-body, short to medium range hybrid transport vehicle demonstrates the feasibility of a hybrid powered aircraft fueled by both lithium ion batteries and jet fuel. Through a series of Class I estimates, including configuration design, weight sizing and sensitivities, determination of performance constraints, fuselage and cockpit design, wing design, empennage design, landing gear design, weight and balance analysis, stability and control analysis, and drag polar estimation, and Class II estimates of the V-n maneuver diagram, and weight and balance, and cost analysis, the early design calculations carefully layout the conversion of a jet aircraft to a hybrid powered aircraft.

This vehicle provides an environmentally friendly form of flight, reducing emissions and dependence on oil. As the number of general aviation aircraft increase and FAA standards call for reduced emissions, designing aircraft with less dependence on oil and fuel is becoming necessary. With battery technology developing, a hybrid aircraft is the perfect transition to greener aircraft for the near future.

Compared to the Boeing 737-100 model, the HTA has comparable weights, with the possibility of a lower takeoff weight. Additionally, with the research behind the SUGAR aircraft,
the availability of hybrid engines would make the transition of power between fuel and electricity much more simple to integrate.

In regards to future work, the battery technology will continue to develop, and, as a result the weight may decrease and the specific energy densities may increase. Additionally, more studies could be performed on the benefits of using fuel and battery power during different phases of the flight. Trade studies could show the advantages and disadvantages of each power source during the phase of flight. The direct operating cost, although preliminary, demonstrates that the batteries, as a power source, are competitive from a cost perspective.

Overall, the hybrid transport aircraft demonstrates strong potential as a preliminary design to meet FAA standards, reduce emissions, and utilize future battery technology to pave the way toward new-age transport jets.
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