

MEANS TO IMPROVE FUEL CELL AIRCRAFT PERFORMANCE THROUGH HYBRIDIZATION AND DESIGN MISSION CONSTRAINTS

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Abstract

The electrification of general aviation serves as the entry-point to making the aviation industry carbon neutral as a whole. The low power requirement and fuel demand of general aviation aircraft makes this segment suitable to explore various configurations of electrical powertrains. Many concepts and retrofit studies of existing aircraft with battery- or fuel cell-electric power source have been developed in the past decade, but these concepts usually result in limited range and performance. One solution to increase the aircraft range is to use the combination of fuel cells and batteries in a hybrid-electric configuration and to optimize the load sharing between them.

In the present study, a simulation model is developed that represents the forces acting on the aircraft as well as the two energy systems, i.e. battery and fuel cell. With this model, critical flight segments such as takeoff and climb-out are analyzed to identify principal design requirements, while considering certification criteria and environmental conditions. A Cessna 400 aircraft is used as a reference. For this aircraft it is demonstrated how to overcome some of the performance limitations due to electrification by developing a power split strategy that leverages the advantage in terms of energy density and flexibility, respectively. The results demonstrate how an adjusted trajectory can reduce the power train weight and therefore keep payload reduction at a minimum.

1. NOMENCLATURE

a	Acceleration
A	Area
c	Coefficient
e	Oswald factor
F	Force
g	Earth gravitational acceleration
I	Current
m	Mass
p	Pressure
P	Power
U	Voltage
T	temperature
v	Airspeed
α	Angle of attack
γ	Flight path angle
η	Efficiency
λ	Stoichiometry coefficient
Λ	Aspect ratio of wing
ρ	Density

2. INTRODUCTION

To achieve the European Union's ambitious targets of reducing the overall climate impact of the aviation sector as outlined in Flightpath 2050 [1], changes in all aircraft sectors are necessary. Replacing the conventional powertrain by an electric alternative is an effective solution to completely avoid local emissions. Retrofitting an existing aircraft can be faster and more economically implemented than developing a new aircraft from the ground up. Starting point will be the general aviation sector, with CS-23 [2]

certification, due to the limited energy density of components at this stage of fuel cell and battery development. In the present analysis a combination of fuel cells and batteries is chosen as energy source. This combines the high-power density of fuel cells and the flexibility and high efficiency of batteries.

This study evaluates, if the retrofit of an existing airframe is possible and which compromises have to be made. It is analyzed, in which way the performance of a fuel-cell based powertrain can be improved by battery-hybridization. Additionally, it is analyzed, which requirements are placed by the flight trajectory and due to the certification process. A simulation model combining the aerodynamic forces acting on the aircraft and a hybrid electric powertrain sizing model is used.

3. METHODS

3.1. Aircraft parameters

As a reference aircraft a Cessna 400 (COL4) is chosen. Important aircraft parameters are taken from the aircrafts pilot operating handbook [3], see also Table 1. The reference design provides a mass budget of engine and fuel of 495 kg for the design range of 2038 km. This mass budget ought to be available for the electrical powertrain, to maintain similar performance. If the electrical powertrain is heavier than that, either the payload mass or the range will have to be reduced to achieve the same power to takeoff mass ratio. Apart from replacing the powertrain no further modification of the aircraft is considered, especially those that would require new fuselage development.

Component	Value	Page in Ref. [3]
Engine power (takeoff)	225 kW	1-4
Engine power (cruise)	195 kW	1-4
Maximum weight	1633 kg	1-5
Rotate Speed	65 KIAS	5-13
Best Rate of Climb Speed	110 KIAS	4-12
Cruise speed	181 KIAS	1-2
Payload	211-500 kg	1-2
Passengers	4	2-12
Fuel	401 l	1-3
Range	2038 km	5-31
Engine	Continental TSIO-550-C	1-3
Propeller	Hartzell 198cm Constant Speed	1-3

Table 1 Aircraft parameters from Pilots Operating Handbook [3]

For modeling the aerodynamic forces and sizing the electrical powertrain a reference profile is needed. A publicly available flight trajectory was chosen, that offered the typical profile with constant climb, cruise and descent phases [4]. The original profile was smoothed and extended by a typical maximum power takeoff phase. The groundspeed values are interpreted as equivalent to true airspeed (TAS), assuming that no wind was present during the original flight. The result is shown in Figure 1.

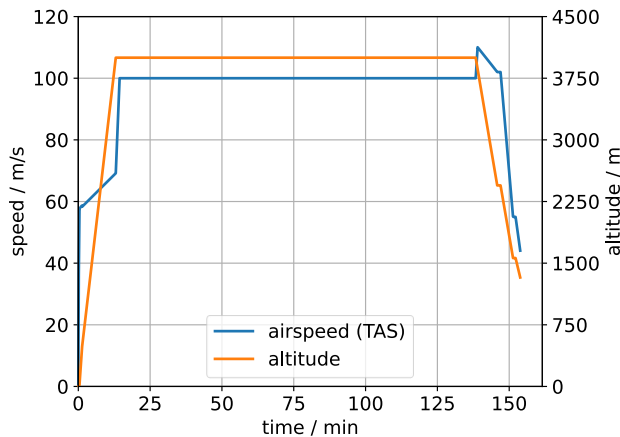


Figure 1 Flight profile [4]

3.2. Aircraft modelling

The aircraft flight performance is modeled by calculating the force equilibrium of the four governing forces acting on it: lift, drag, thrust and weight. The relation between them is illustrated in Figure 2 and Equation (1), (2) according to their orientation in the 2D-plane. The original concept of the aircraft modelling was taken and adapted from [5]. Equation 1 balances all forces in thrust axis, Equation 2 in lift axis. All forces are assumed to act on the center of gravity to eliminate the need for torque and stability considerations. For steady flight, all forces are in equilibrium and acceleration is zero. For timespans with changes in airspeed, constant acceleration between neighboring timesteps is considered. The force equations are taken from [6] and are described in Equations (3)-(5) and Table 2. The equilibrium is solved for the angle of attack α and the power P at each timestep using SciPy's root solver [7].

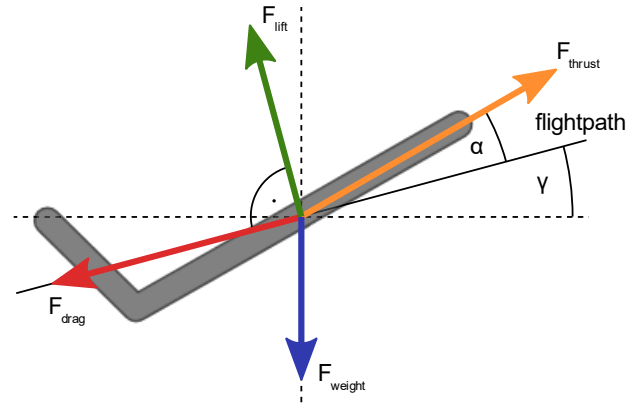


Figure 2 Equilibrium of aerodynamic forces

$$F_{thrust} - F_{drag} \cos(\alpha) - F_{weight} \sin(\alpha + \gamma) + F_{lift} \sin(\alpha) = ma \quad (1)$$

$$F_{lift} - F_{weight} \cos(\gamma) + F_{thrust} \sin(\alpha) = 0 \quad (2)$$

$$F_{lift} = c_l(\alpha) \frac{\Lambda}{\Lambda + 2} \frac{\rho}{2} v^2 A_{wing} \quad (3)$$

$$F_{drag} = \frac{\rho}{2} v^2 c_{d0} A_{front} + F_{lift} c_l(\alpha) \frac{\Lambda}{\Lambda + 2} / (\pi e \Lambda) \quad (4)$$

$$F_{thrust} = \eta_{Prop} \frac{P}{v} \quad (5)$$

$$F_{weight} = m g \quad (6)$$

Sym.	Variable	Value	Ref
α	angle of attack	min: -10° , max: 13°	
γ	flight path angle		
a	acceleration		
v	true airspeed		
g	earth grav. acceleration	9.81 m/s ²	
m	mass	1633 kg	[3]
P	power	max: 225 kW	[3]
ρ	air density		[8]
η_{Prop}	propeller efficiency	0.8	[6]
c_l	lift coefficient		[9]
A_{wing}	wing area	13.1 m ²	[3]
A_{front}	front area	4.65 m ²	[3]
Λ	aspect ratio	9.07	[4]
e	Oswald factor	0.75	
c_{d0}	zero lift drag coefficient	0.0055·4	[6]

Table 2 Variables and constants occurring in the equilibrium of aerodynamic forces and their values

For the aircraft modeling the following assumptions apply:

- All forces act on the center of gravity
- Engine thrust is in the same axis as wing angle of attack
- Engine efficiency is modelled by propeller efficiency only (not considering nacelle/Mach drag)
- Flaps and control surfaces are neglected, lift coefficient $c_l(\alpha)$ from standard configuration without flap deflection is used in all cases
- International Standard Atmosphere (ISA) conditions [8] are applied (except for takeoff and climb gradient considerations)

3.3. Fuel cell and battery modeling

The resulting power profile of the aircraft simulation is used to size the electric powertrain, see Figure 3. It consists of electric motor and power electronics to convert the energy stored in the hybrid fuel cell system to propulsive power. E-motor, inverter and converter are modelled through fixed power densities and efficiencies, see Table 3. The hybrid fuel cell/battery system is modelled in greater detail with the model already published in [10], [11], [12]. Firstly, the load distribution between fuel cell system (FCS) and battery is set by defining a minimum and maximum power output of the FCS. For phases where the required power is higher than the maximum FCS-power, the batteries are assisting. When the required power is lower than the minimum FCS-power, the excess power may be used to charge the battery. This recuperates some power back during low-power phases and makes sure the FCS is operated in an efficient operating point. The size and number of fuel cell stacks is determined by the maximum allowable current density, with the balance of plant (BoP)-components necessary to operate the fuel cell stack (FC-stack) parametrized according to Table 3. The overall system is sized component by component based on the required power and resulting mass flows required to operate the FC-stack.

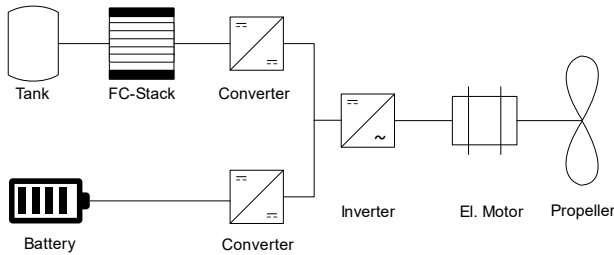


Figure 3 Electrical propulsion system model [12]

Comp.	Modeling	Assumptions	Reference
FC-Stack	Model fitted with measurement data [10]	$T_{in} = 85^\circ\text{C}$, $\lambda_{Cath.} = 2$, $\lambda_{Anode} = 1.05$	Powercell P2
Com-pressor	Analytic with const. eff. and mass/mdot ratio	$\eta_{isentropic} = 0.8$, pressure ratio $_{max} = 2.6$	Fischer EMTC120k
Humidi-fier	Analytic with const. mass/mass flow-ratio		Fumatech Ecomate H100N
Heat exch.	Empirical correlation [11]		louvered fin design
Cooling pump	Analytic with const. eff. and mass/volume flow-ratio	$\eta_{overall} = 0.57$	DS600-24A Type 6902
Tank	Gravimetric efficiency	0.16 kg H ₂ /kg tank, 10 % H ₂ reserve	liquid hydrogen tank [13]
Battery	Equivalent circuit model with OCV measurem. [12]	90 % DoD, SoH = 1, $U_{nom} = 800\text{V}$	LG18650HG2
Power electronics	Fixed power densities	$\eta_{electric} = 0.924$, $P/m_{em} = 5.2$ kW/kg, $P/m_{conv} = 21.5$ kW/kg	[14], [15]

Table 3 Component modeling and reference

Component masses are scaled linearly from their reference. During the sizing process the internal BoP-power requirement is provided by the FCS-stack and considered through calculating and accounting for the ratio between FC-stack-power and BoP-power requirement for each timestep. The optimal values for the stack pressure and stack relative humidity are identified and applied based on the overall system efficiency. The battery is sized by the required nominal voltage of 800 V, and the more demanding of the two requirements total mission energy and peak battery power. Piping and wiring is neglected, no additional mass is considered neither for the reference powertrain nor for the electrical replacement.

4. RESULTS AND DISCUSSION

With the two models, i.e. aircraft model and fuel cell/battery hybrid model, an electrical propulsion system is sized starting from the specified mission profile over the required engine power to the sizing of each individual component. The required power for the baseline mission profile (see Figure 1) was calculated using the model and assumptions of the previous chapters, see Figure 4. During the takeoff phase the power is limited to the maximum engine shaft power of 225 kW. After rotate speed is achieved, the aircraft ascends to an altitude of 15 m and accelerates to the best rate of climb speed (57 m/s) at the same time. Then the first climb segment is performed with constant airspeed and maximum power to an altitude of 457 m according to CS-23.2120 [2]. After that, the climb rate is reduced to 5 m/s like in the reference mission profile in Figure 1. Therefore, the required power reduces to 150 kW. The second climb segment is performed with constant indicated airspeed, therefore the groundspeed and the required power increases quadratic with the decreasing air density. After reaching the cruise altitude of 4000 m, the aircraft is accelerated to its cruise speed of 100 m/s. After around two hours the aircraft descends to the destination altitude of 1326 m, doing two short holding patterns with constant altitude in between.

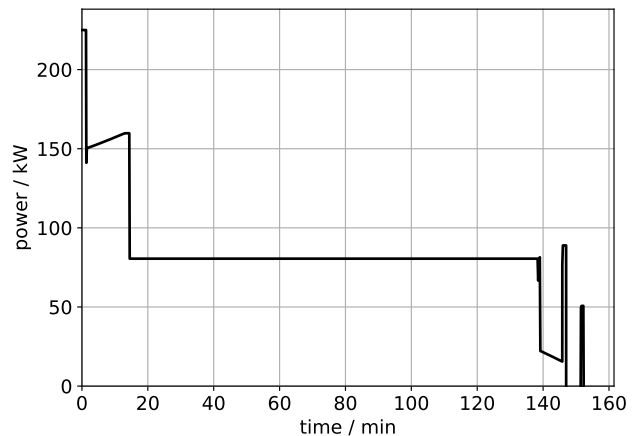


Figure 4 Required engine shaft power for the baseline mission profile

4.1. Propulsion system mass and volume comparison

With the power profile shown in Figure 4, the hybrid fuel cell system is sized. Figure 5 shows the mass of the hybrid fuel

cell system (left) compared to the reference design (right). The comparison is based on a maximum FCS-power of 100 kW and a maximum current density of 0.8 A/cm^2 . These were found to be the optimal values for this power profile and are used for all studies. The reference fuel mass was reduced to 50 % of the maximum capacity, in accordance with the shorter baseline mission range. This leads to a cruise efficiency of 45 % for the FCS, and an overall hydrogen consumption of 14.1 kg. Two FC-Stacks á 368 cells and 30 Ah battery capacity are necessary. It can be seen that the overall propulsion system mass rises by 75 kg compared to the reference powertrain, or in other words, the payload mass has to be reduced in the order of one passenger with state-of-the-art technology for this retrofit. Similar results can be found when the cruise duration is extended to the design range (2038 km) with a 56 kg decrease in payload, see Figure 6. The overall tank volume is calculated to be 1.5 m^3 for the design range. Assuming a spherical tank, it's radius would be 1.42 m, which would be wider than the main fuselage width. Therefore, a cylindrical tank would be easier to integrate in the existing fuselage, but this will surely decrease the payload capacity further.

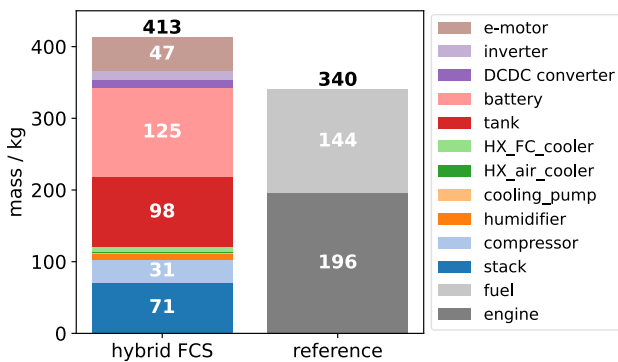


Figure 5 Mass comparison of the hybrid electric propulsion system with the conventional reference for the baseline profile according Figure 1.

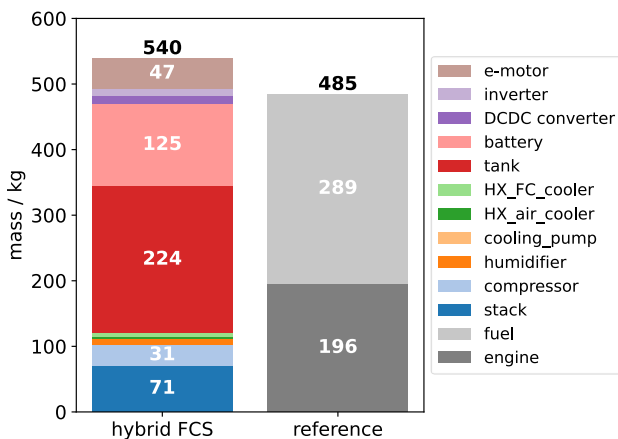


Figure 6 Mass comparison of the hybrid electric propulsion system with the conventional reference for the design range

4.2. Analysis of different degrees of hybridization

In this analysis the load distribution between FCS and battery is optimized for the design range, i.e. 2038 km. To achieve 225 kW engine power, varying power shares between fuel cell system and battery may be defined (respecting efficiencies of electric motor, inverter and

converter). The FCS-power share is varied in 50 kW increments up to the maximum engine power and the resulting mass is compared, see Figure 7 and Figure 8 respectively. 250 kW FCS-power is equivalent to a FCS-only configuration without battery. The total mass is almost constant for the configurations with 100 kW – 250 kW FCS-power, see Figure 8. The savings on battery mass are traded with additional FC-stack- and BoP-mass. Interestingly the tank mass does not increase with rising FCS-power shares. Since the FCS has to be sized for the maximum power during takeoff, a configuration with higher FCS-power shares has a lower load factor during the cruise phase. Since all configurations are sized with the same maximum current density during takeoff, this leads to lower current densities during the cruise phase for FCS-dominated configurations and therefore higher efficiency. Accumulated over the extensive cruise phase this leads to lower overall hydrogen consumption, even for the FCS-only configuration. In the 50-kW variant, the FCS-power is not sufficient to maintain the cruise power on its own, therefore the battery is active during the whole mission and the necessary battery energy increases drastically to 245 kWh. The battery only configuration is not plotted for scaling reasons. However, the overall system mass would result in 2860 kg with 547 kWh battery energy.

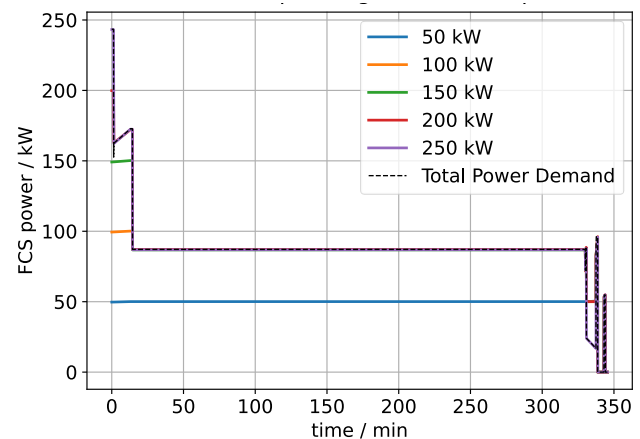


Figure 7 Load profile during the design range profile for different FCS power shares

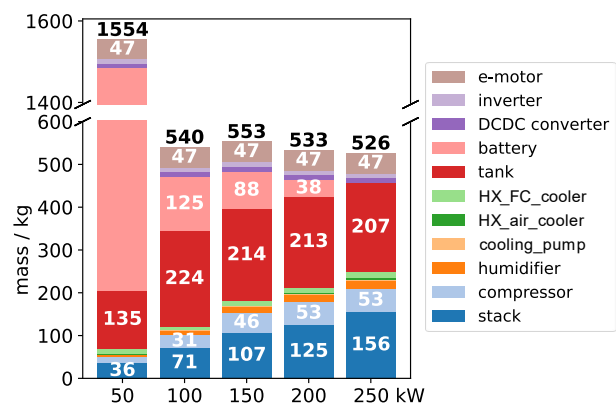


Figure 8 Mass of the hybrid electric propulsion system for different FCS power shares

4.3. Analysis of the required ground roll for varying engine power

CS-23.2115 requires the determination of the length of the ground roll, meaning the distance traveled on the runway

until takeoff. Since this is highly dependent on the available power, this was analyzed in detail for the maximum takeoff mass 1633 kg. The overall engine power was varied in 25 kW increments, independently from the underlying architecture. For this study, the lift part of the force equilibrium was neglected and a rolling resistance with a friction coefficient of 0.018 was added, assuming a concrete runway according to [16]. The density altitude was set to 4260 m, according to CS-23.2105. this can be assumed to be the worst-case conditions for the aircraft to operate in. The minimum airspeed for calculating propeller thrust was set to 25 m/s. With this the possible acceleration with maximum power was determined and the distance traveled until reaching the required rotate speed of 33.5 m/s was calculated. The resulting ground roll distance is shown in Figure 9. It depends roughly quadratically on the available power (fit curve: $2.126e-8x^2-0.011x+1646$). The calculated ground roll for the reference 225 kW variant (orange bar) of 280 m is still considerably shorter than the 670 m stated in the pilots operating handbook for the same density altitude. This can be explained in part by conservative information provided in the handbook, but probably also hints at limitations in the model representation due to the constant thrust efficiency and drag assumptions. After all it can be found, that by increasing the allowable runway length the necessary power could be drastically reduced, if no limitations during other segments occur. Additionally, implementation of a high-power battery for a quick power boost could be used to temporally increase the available power, without having to oversize the entire FCS.

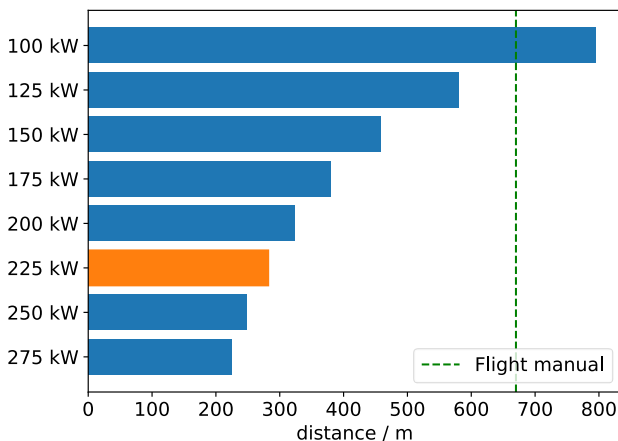


Figure 9 Takeoff acceleration distance depending on the available engine power

4.4. Analysis of the climb gradient for varying engine power

Here, the first climb segment to 457 m is analyzed in detail. CS-23.2120 requires a minimum climb gradient of 8.3 %, up to a density altitude of 4260 m (CS-23.2105). The indicated climb speed was set to the best rate of climb speed (57 m/s). The achievable climb gradients depending on available power were calculated for power values varying in 25 kW increments. The results are shown in Figure 10. It can be seen, that the maximum climb gradient linearly depends on the available power. The installed power could be reduced to 175 kW, and still suffice the CS-23 requirement for maximum takeoff weight. This would surely reduce the maximum allowed cruise altitude, but might be a better approach than compromising payload capacity or range for the given airframe.

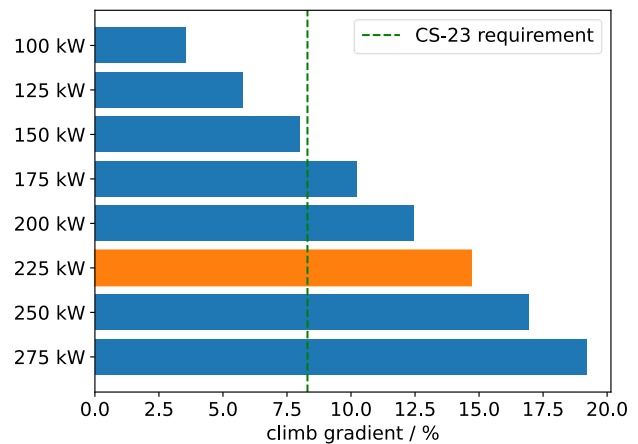


Figure 10 Climb gradient achievable depending on available engine power

4.5. Propulsion system mass for varying engine power installed

Taking the results from the analysis of ground roll and climb gradient, it can be deduced that it is possible to reduce the installed engine power to 175 kW without compromising the overall aircraft performance capabilities drastically. This helps with reducing the overall propulsion system mass and regains payload mass. To analyze this, the required power during the takeoff phase and first climb segment was reduced in 25 kW increments in the range possible with regard to section 4.3 and 4.4. The resulting propulsion system mass is shown in Figure 11. Only reducing the required power neglects that the time needed to reach the 457 m altitude requirement increases, but an increase in the order of 10-30 seconds of time in the climb phase time doesn't affect the overall mission energy significantly. The FCS power is kept constant at 100 kW for all variants, so that the change in power only affects the battery and power electronics. Reducing from 225 kW to 200 kW the number of battery cells in parallel can be reduced from 10 to 9 cells, reducing the overall battery mass by 12 kg. Since the active phase of the battery is relatively short, the number of battery cells in parallel is affected by the maximum power one cell can deliver. For 225 kW and 250 kW the determining sizing criterium is the maximum power of one cell, whereas for 175 kW and 200 kW the overall battery capacity is decisive. This demonstrates that selecting an appropriate battery cell type for the selected battery hybridization strategy is crucial.

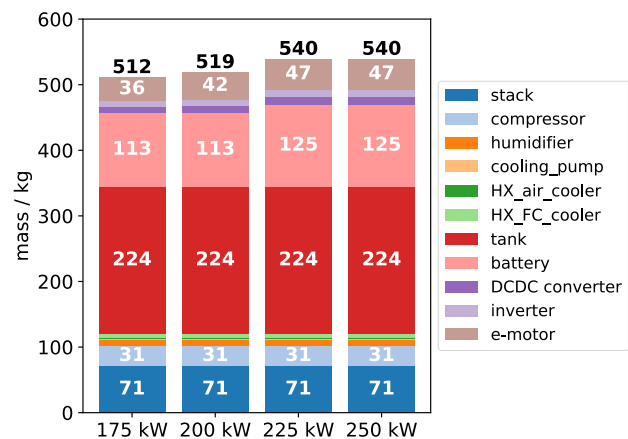


Figure 11 Mass of the hybrid electric propulsion system depending on available engine power

5. CONCLUSION

This study shows, that replacing the conventional powertrain by an electric equivalent is possible for a Cessna 400 aircraft, but not without compromises. In general, the heavier propulsion system and additional tank integration costs about one passenger in payload capacity or equivalent range. The integration of the tank in the existing fuselage is an additional challenge. Contrary to aviation gas or kerosene it is difficult to store the tank in the wings, therefore additional space for a cylindrical fuel tank in the main fuselage needs to be allocated. Hybridization doesn't show a significant improvement when looking at overall propulsion system mass alone, but could be used as a flexible way of fulfilling high takeoff- and first climb segment-power requirements, especially at high outside temperatures.

The optimal design depends on the design range, maximum required power and technology improvements of all included components. Using a high-power battery could help fit the design for different ground roll and environment conditions. Alternatively, it is possible to reduce the installed system power by carefully revisiting the aircraft performance requirements regarding ground roll, climb gradient and maximum altitude. This way compromises could be made by reducing the performance characteristics determined by excess power slightly, in favor of general performance metrics like i.e. payload and range.

6. ACKNOWLEDGMENTS

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