SIZING STUDIES OF LIGHT AIRCRAFT WITH PARALLEL HYBRID PROPULSION SYSTEMS

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Abstract

In the past decades, interest in cleaner, alternative propulsion increased rapidly. Accordingly, novel approaches for aircraft propulsion are required to meet the spirit of the time. This paper aims to assess the viability of parallel hybrid-electric propulsion systems for General Aviation (GA) aircraft with state of the art technology. Therefore, a comparison to the conventionally powered competitors is performed by using initial sizing results. An assessment of clean sheet designs is performed, as well as an evaluation of retrofit designs. Additionally, parameter studies on battery specific energy and aerodynamic efficiency are conducted to analyze their potential in improving viability and advantages of such aircraft. The studies' results indicate that parallel hybrid-electric light aircraft are able to offer considerable benefits over conventional aircraft, even with today's technological standards. However, several design aspects have to be taken into account to make full use of the benefits arising from these concepts.

Nomenclature

acc = Acceleration
alt = Altitude
AR = Wing Aspect Ratio
bat = Battery
B_s = Breguet Range Factor
BSFC = Brake Specific Fuel Consumption
c_D = Drag Coefficient
c_L = Lift Coefficient
conv = Conventional
CS = Certification Specifications
CTOL = Conventional Take-off and Landing
E = Energy
e = Oswald Efficiency Factor
E^* = Mass Specific Energy
EM = Electric Motor
GA = General Aviation
h = Height / Altitude
H_E = Degree of Hybridization of Energy
H_P = Degree of Hybridization of Power
i = Flight Phase
ICE = Internal Combustion Engine
LFC = Laminar Flow Control
m = Mass
MSL = Mean Sea Level
MTOM = Maximum Take-off Mass
nc = Non-consumable
opt = Optimum
P = Power
P^* = Mass Specific Power
P/W = Power-to-Weight Ratio
PH = Parallel Hybrid
prim = Primary
s = Range
S = Wing Area
STOL = Short Take-off and Landing
TOD = Take-off Distance
v = Velocity
VTOL = Vertical Take-off and Landing
W/S = Wing Loading
ε = Stop Criterion
η = Efficiency

1. INTRODUCTION

The use of consumable energy sources, i.e. the combustion of fuel, dominates aircraft propulsion since the beginning of aviation history. To ensure a future of sustainable aviation, innovative and efficient propulsion concepts must be considered. Fully electric propulsion systems are not yet capable of replacing conventional, fuel burning systems. This is caused mainly by high battery weight resulting from the low specific energy of currently available batteries compared to fossil fuel [1, 2]. Thus, fully electric flight missions are restricted to extremely short ranges, currently. The advantages that electric propulsion entails, though, are a huge opportunity to make aviation more efficient and eco-friendlier. Electric powertrains offer high energy conversion efficiencies in comparison to Carnot-efficiency restricted fuel burning engines, as well as low emissions and low noise pollution. A hybrid-electric propulsion system, including both a consumable and a non-consumable energy carrier, is able to combine the advantages of fuel-based systems and battery powered systems. Parallel hybrid propulsion systems, in particular, offer the possibility of simple retrofit designs, which today appear to be the easiest way to ensure a large number of low-emission aircraft. Besides, the direct connection of the combustion engine to the propeller shaft of a parallel hybrid powertrain (see Figure 1) offers major efficiency benefits in comparison to a serial hybrid configuration [3]. This leads to the investigation of the parallel hybrid design being an inevitable step to accomplish today's aspiring emission targets [4].

This paper aims to investigate the design space of parallel hybrid-electric General Aviation (GA) aircraft. Within the scope of initial sizing, the viability of such aircraft with state of the art technology is assessed in comparison to their conventionally powered counterparts. The whole design space is searched for the optimum design point and the optimal degree of hybridization with respect to the design objective of a minimum take-off mass, while mission requirements (e.g. range, payload) are varied. The comparison of each hybrid aircraft to the equivalent conventional aircraft with the same mission requirements is conducted using the assessment criterion of maximum

take-off mass (MTOM). Consequently, aircraft costs are regarded too, since cost scales almost linearly with aircraft weight for this aircraft class [5, 6]. Moreover, the required fuel mass is considered, as well as the primary energy use. Clean sheet designs are compared, which feature a completely new design that is perfectly fitted to the hybrid propulsion system in terms of e.g. wing area. Their design point changes compared to a conventional aircraft operating at the same mission. Also, retrofit designs are regarded, where the propulsion system is replaced, while wing area and design point stay identical.
Furthermore, parameter studies for varying technology levels of battery specific energy and aerodynamic efficiency are performed to examine their capability of enhancing the benefits of parallel hybrid-electric powered GA aircraft.

This paper is structured the following way: Following this introduction, the classical methodology for initial sizing is briefly described in section 2. The methodology of a new approach for hybrid-electric aircraft is also presented. In section 3, the new method is applied as the design space of parallel hybrid-electric GA aircraft is explored for varying parameter ranges. The results are evaluated with respect to the viability of such aircraft. Finally, section 4 gives a comprehensive conclusion.

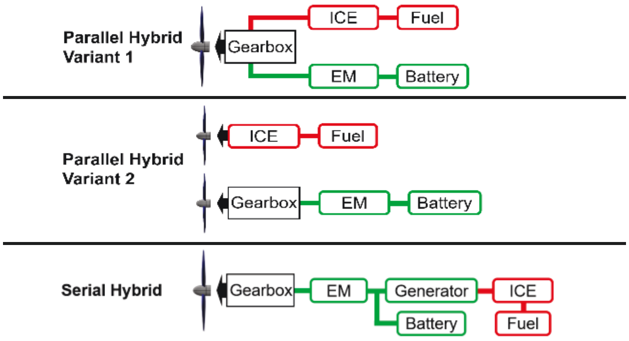

Figure 1 - Hybrid-Electric Powertrains

2. INITIAL SIZING

Initial sizing is the "*most important calculation in aircraft design*" [7]. This is due to the fact that the rough dimensions of a new aircraft are determined during this process, just by considering a hand full of requirements and specifications. These dimensions are specified as the MTOM, the maximum required power (P), and the wing area (S). If miscalculations appear in the course of initial sizing, the new aircraft will most likely not meet its requirements. During sizing, numerous aircraft configurations are analyzed to provide the most important information for the choice of the final configuration.

2.1. Requirements

The general aircraft design process always starts with the definition of the requirements that a new aircraft shall fulfill. Requirements needed for the aircraft sizing are classified in customer and authority's requirements. Typical customer requirements are presented below [8]:

- Payload Mass (crew, passengers, baggage)
- Cruise Speed, Range and Altitude
- Rate of Climb
- Take-Off Ground Run
- Turn Performance (maximum bank angle)

Authority's requirements are dependent on the class of aircraft. Within the scope of this paper, the CS-23 and its categories of Normal, Utility, Aerobatic and Commuter are important. This is due to the algorithm, which was used for the presented studies on parallel hybrid aircraft, being mainly developed for GA aircraft. In terms of initial sizing, representative requirements given by certification rules are:

- Stall Speed
- Take-Off and Landing Speed
- Obstacle Heights (for take-off and landing)
- Rate of Climb at Service Ceiling

Once all requirements are known, the actual aircraft sizing can begin.

2.2. Point Performance

The classical sizing itself is split into two major parts – the Point Performance and the Mission Performance.
During Point Performance calculations, preliminary work is done to determine the required maximum power and the wing area. In this first step of the initial sizing, these parameters are computed in dependence on the MTOM as the power-to-weight ratio (P/W) and wing- or surface loading (W/S). The MTOM is calculated during the Mission Performance afterwards. Some authors (e.g. Roskam [9]) utilize power loading instead of P/W, which represents the inverse value and mirrors the constraint diagram about the x-axis. To identify the suitable values for the above-mentioned ratios, a so-called constraint analysis has to be performed. In the framework of this analysis, the necessary P/W to fulfill top level performance requirements is determined as function of the W/S. For the used sizing tool, constraint functions based upon the equations given by Gudmundsson [8] have been implemented [10]. To proceed the analysis, these constraints are drawn as curves into the constraint diagram (see Figure 2), which relates the W/S to the P/W and thus, depicts the design space of an aircraft.

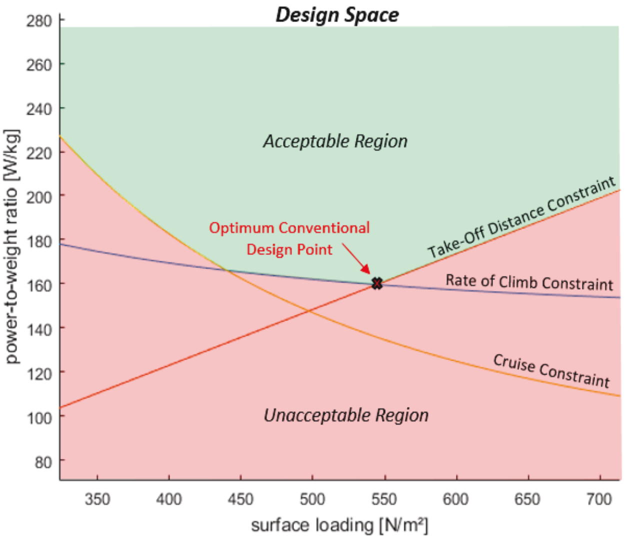

Figure 2 - Typical Constraint Diagram

In consideration of a sufficient power supply in all flight attitudes where those performance requirements must be met, the constraint diagram is divided into an acceptable and an unacceptable region. While in the unacceptable region, every design point with a defined P/W and a W/S is below the highest constraint at the current W/S, all points in the acceptable region are above the highest constraint. This means that aircraft which are designed with values from the unacceptable region will not fulfill at least one performance requirement due to a lack of power. In contrast, aircraft designed with values from the acceptable region will fulfill all top level performance requirements that have been taken into account. Yet, they may be oversized in terms of power if not designed to the exact P/W requirements defined by the highest constraint at the current W/S.

A conventional aircraft usually has one set of wings with a specific wing area, as well as it only has one propulsion system producing a certain amount of power. Thus, it is indispensable that just one design point is chosen from the acceptable region of the design space. To make a decision, the so-called design objectives for a new aircraft must be known. They highly influence the choice of the design point by defining additional design requirements. For GA aircraft certified according to CS-23, the design objectives are most commonly minimum costs and therefore a light weighing aircraft, combined with a small size. With respect to these objectives, the optimum design point of conventionally powered aircraft can always be determined pursuant to a simple rule:

> *“At first priority, the lowest possible P/W is chosen, as a lower P/W leads to smaller and lighter engines and thus lower cost. At second priority, the highest possible W/S is selected, since a higher W/S leads to lower structural weight, less wetted area and thus lower drag.”* [11]

The resulting optimum design point, which has to be chosen for a new aircraft in regard to the constraint functions, is exemplarily shown in Figure 2. This design point is only valid for conventional aircraft, which are powered by propulsion systems based on consumable energy sources like jet engines or internal combustion engines (ICE). The analysis of the design space for unconventional propulsion aircraft is a central aspect of this paper.

2.3. Mission Performance

The aim of the Mission Performance is to estimate the MTOM of a new aircraft, while the ratios of P/W and W/S are already known from the Point Performance. Using the resulting MTOM, the total values for maximum power and wing area can be calculated. In general, the MTOM is composed of the mass proportions given in Equation (1).

$$MTOM = m_{empty} + m_{fuel} + m_{payload} \tag{1}$$

The payload mass is known, since it is an important customer requirement. However, the empty mass, including the aircraft’s structure, engines, landing gear, fixed equipment, avionics and other systems, as well as the fuel mass have to be calculated. Both masses are dependent on the MTOM, so that an iterative process must be conducted, which starts with a guess of the MTOM (see Figure 3).

For any currently regarded MTOM, the empty mass is given by statistics as an empty mass fraction. These statistics rely on already existing aircraft and relate the empty mass fraction to the MTOM. They vary for different types of configurations, different classes of aircraft with different missions, different years of construction and even for different authors offering these statistics. Therefore, the choice of the proper statistic is very important.

During the classical sizing, the fuel mass for conventional aircraft is calculated using so-called fuel fractions, which represent the ratio of the total aircraft mass at the end of a flight phase to the mass at the start of a flight phase (see Equation (2). The difference between both masses is the mass of the burned fuel.

$$\frac{m_i}{m_{i-1}} = \frac{m_{end}}{m_{start}} \tag{2}$$

These fuel fractions are given by statistics for all flight phases of a mission besides the cruise and loiter phases.

For the cruise and loiter phases, the corresponding fuel fractions are normally computed using the Breguet Range Equation [8, 12].

$$\frac{m_i}{m_{i-1}} = e^{-\frac{s}{B_s}} \tag{3}$$

The ratio between the total required fuel mass and the MTOM, which is referred to as the total fuel fraction, is calculated according to Equation (4).

$$\frac{m_{fuel}}{MTOM} = \left(1 - \prod \frac{m_i}{m_{i-1}}\right) \tag{4}$$

Once the empty mass fraction and the total fuel fraction are determined for the current MTOM, the new MTOM is computed using the following formula [13], which is a transformation of Equation (1).

$$MTOM = \frac{m_{payload}}{1 - \frac{m_{empty}}{MTOM} - \frac{m_{fuel}}{MTOM}} \tag{5}$$

In regard to a stop criterion that is specified as a difference in mass, the difference between the old and the new MTOM is calculated. If the stop criterion is not fulfilled, the next iteration loop of the Mission Performance process is performed. If it is fulfilled, the new MTOM corresponds to the final MTOM that is used to determine the thrust, respectively the power and the wing area of the new aircraft.

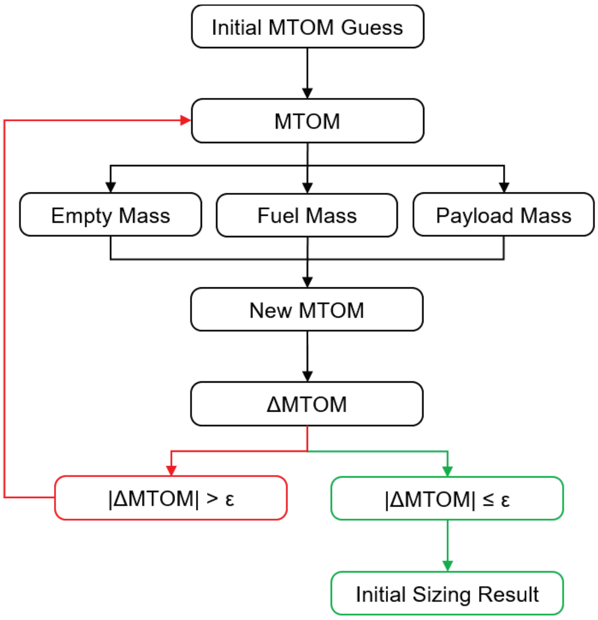

Figure 3 - Classical Mission Performance Sizing

The presented approach for the calculation of the fuel mass is, just like the approach for the determination of the optimum design point, not applicable to unconventional propulsion aircraft. This is due to the possible use of propulsion systems based on non-consumable energy sources like batteries, which do not or just partially lead to a loss of mass during the flight mission of an aircraft.
Thus, a new approach has been developed that serves the purposes of this paper.

2.4. Methodology of a New Approach

The algorithm that is used to perform the studies included in this paper was developed to deal with hybrid-electric aircraft. It is able to identify the optimal design point of both parallel hybrid and serial hybrid aircraft as well as the corresponding degree of hybridization. Besides the hybrid-electric sector, it also covers purely conventional propulsion and all-electric propulsion.

The methodology of the algorithm is, similar to the classical sizing methodology, separated into Point Performance and Mission Performance.
During Point Performance calculations, the optimal design point is determined with respect to a defined design objective (e.g. MTOM, primary energy). Therefore, a large number of design points with different degrees of hybridization are analyzed in the Mission Performance to evaluate each point. This is due to the fact that values for design objectives are calculated during mission analysis.
In order to prepare these Mission Performance calculations, the algorithm divides the overall power demand arising from the design point in the constraint diagram (see Figure 4) to the electric motors (EM) and the ICEs during Point Performance. This subdivision is done for different degrees of hybridization of power H_P and for every potential design point. This leads to the definition of so-called split points, as exemplarily presented in Figure 4. The H_P refers to the installed power of the propulsion devices and is defined by Equation (6) for parallel hybrid-electric propulsion systems.

$$H_{P,PH} = \frac{P_{EM,max}}{P_{max}} \tag{6}$$

Simultaneously, the amount of energy obtained from batteries and fuel in every time step of a flight phase is identified by a degree of hybridization of energy H_E. It is determined for each flight phase during Point Performance by relating the vertical position of a split point to the P/W-value of the constraint that corresponds to flight phase.

$$H_{E,i} = \frac{\Delta E_{nc}}{\Delta E} \tag{7}$$

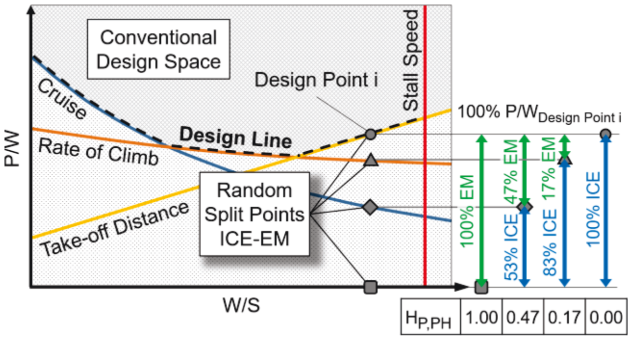

Figure 4 - Design Space of Parallel Hybrid Aircraft [10]

During Mission Performance sizing, aircraft dimensions and masses are computed based on the classical iterative process shown in Figure 3. However, the estimation of the energy carrier masses does not rely on fractions as for the conventional sizing process, so that the process had to be slightly adapted in relation to Figure 3. This is due the mix of consumable (carbon-based fuels) and non-consumable (batteries) energy sources, where the non-consumable energy sources exhibit a constant mass during flight. For certain battery types (e.g. Lithium-air) it is even possible that the mass of the battery grows as it is discharged [14]. Thus, a universally valid, energy-based approach is applied to determine required energy carrier masses.
For the new approach of mass calculation, the transport energy demands in small time steps of each flight phase are computed. Energy demands can arise from aerodynamic drag, acceleration (kinetic energy), altitude change (potential energy) and ground friction.

$$\Delta E = \Delta E_{drag} + \Delta E_{acc} + \Delta E_{alt} + \Delta E_{ground} \tag{8}$$

By relating the energy and the time of a time step, the required power is calculated. Merged with values for brake specific fuel consumption (BSFC) of the ICE, battery specific energy and information with respect to the efficiency chains, the energy carrier masses can be determined for every time step.
Consequently, the whole flight mission is simulated for every iteration step until the stop criterion is met.

Besides MTOM, the use of the energy-based method allows for the evaluation of an aircraft during Point Performance in terms of primary energy.
The primary energy is a measure for total energy that was harvested directly from natural resources to provide an amount of energy to the consumer. E.g. fossil fuel has to be refined from raw oil, which prior was extracted from the soil. All of the used energy to produce the fuel is summed up in a primary energy factor. The factors in Germany from 2016 are 1.1 for fossil carbon based fuel and 2.8 for electricity [15]. The factor for electricity is that high because of the current composition of electricity, where coal-burning and nuclear power plants have a big share. The factor most likely will decrease as the use of renewable energy sources increases.

A more detailed description of the methodology of this sizing approach for hybrid-electric light aircraft is provided by Finger [10], Ludowicy [16] and Rings [17]. Applications of the method were performed for GA aircraft in conventional take-off and landing (CTOL) and vertical take-off and landing (VTOL) configuration [3, 18].

3. VIABILITY STUDIES

In the following chapter, the viability of parallel hybrid-electric powered GA aircraft is evaluated for clean sheet designs as well as for retrofit designs. Therefore, a baseline is created to define mission requirements and determine comparison data.
Particular mission parameters are varied to investigate their influence on the viability of parallel hybrid GA aircraft featuring a clean sheet design. Accordingly, a mission profile that suites best for such aircraft is identified.
In order to evaluate the feasibility of retrofitted aircraft, different degrees of hybridization are analyzed and compared to the baseline aircraft.

Besides, parameter studies are performed on clean sheet designs to assess the capability of advanced technology levels in enhancing the benefits of parallel hybrid-electric powered GA aircraft.

3.1. Baseline Aircraft

The baseline aircraft serves as a starting point for all studies conducted within the scope of this paper. It combines all requirements, as well as aerodynamic and technological properties that are consulted for the analyses. For all studies, certain parameters are varied, while the remaining parameters equal those of the baseline aircraft.
Ludowicy uses the same baseline for studies of light aircraft with serial hybrid propulsion systems [19].

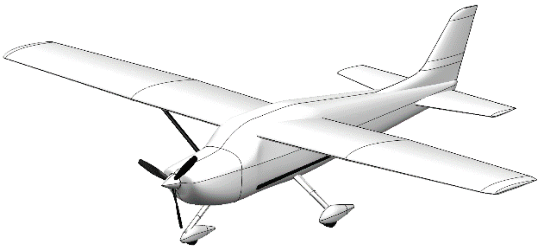

Figure 5 - Conventional Baseline Aircraft

The baseline aircraft is a single-engine four-seat GA aircraft similar to the Cessna 172, which shall be able to carry a payload of 300 kg over a range of 1000 km. During cruise, it is intended to operate at an altitude of 2500 m with a velocity of 55 m/s, where its parasitic drag coefficient ($c_{D,min}$) shall conform to 300 drag counts. The aircraft has a constant speed propeller with a fixed efficiency of 0.8 and shall be able to take off within a ground run of 200 m.
The sequence of the reference aircraft's flight mission is given as taxi, take-off, climb, cruise, loiter, descent and again taxi. The most important aircraft parameters are summarized in Table 1.

Table 1 - Top Level Requirements and Aircraft Data

Requirements	
Payload at Design Range [kg]	300
Design Range [km]	1000
Take-off Ground Run [m]	200
Rate of Climb at MSL [m/s]	4
Stall Speed [m/s]	25
Cruise Speed [m/s]	55
Cruise Altitude [m]	2500
Aerodynamics	
Maximum Lift Coefficient ($c_{L,max}$) [-]	1.8
Parasitic Drag Coefficient ($c_{D,min}$) [-]	0.03
Lift Coefficient at minimum Drag ($c_{L,0}$) [-]	0.25
Wing Aspect Ratio (AR) [-]	7.5
Oswald Efficiency Factor (e) [-]	0.75
Technology	
ICE Specific Power (P_{ICE}*) [kW/kg]	1
ICE Specific Consumption ($BSFC_{min}$) [g/kW/h]	350
E-Motor Specific Power (P_{EM}*) [kW/kg]	5
Battery Specific Energy (E^*_{bat}) [Wh/kg]	250
Propeller efficiency ($\eta_{Propeller}$) [-]	0.8

Following the design rule for conventionally powered aircraft (see section 2.2), the baseline aircraft was sized once to the conventional design point. Additionally, a sizing concerning a parallel hybrid design of the baseline aircraft was conducted for the same mission requirements, while the design point that offered the lowest MTOM was chosen. The design to the lowest MTOM will be referred to as optimal, hereinafter. The sizing results are stated below.

Table 2 - Baseline Aircraft Sizing Results

Payload [kg]	***Conventional***	***Optimal***
MTOM [kg]	1090	1075
W/S [N/m^2]	492	689
P/W [W/kg]	84	133
H_P [-]	0.000	0.535
E_{prim} [MJ]	6772	5168
m_{fuel} [kg]	143.0	107.4

Both designs are highlighted in Figure 6, where the MTOM distribution in the constraint diagram is displayed. The distribution arises from the Point Performance sizing and the mass calculation for numerous split points that are included in the design space, as described in section 2.4.
While for the conventional layout the design point is indicated, the parallel hybrid design is represented by its split point. The split point illustrates the share of EM and ICE in overall required power, analogue to the procedure shown in Figure 4. The design point of the parallel hybrid reference aircraft, though, is solely the vertical projection of the split point to the highest constraint curve.

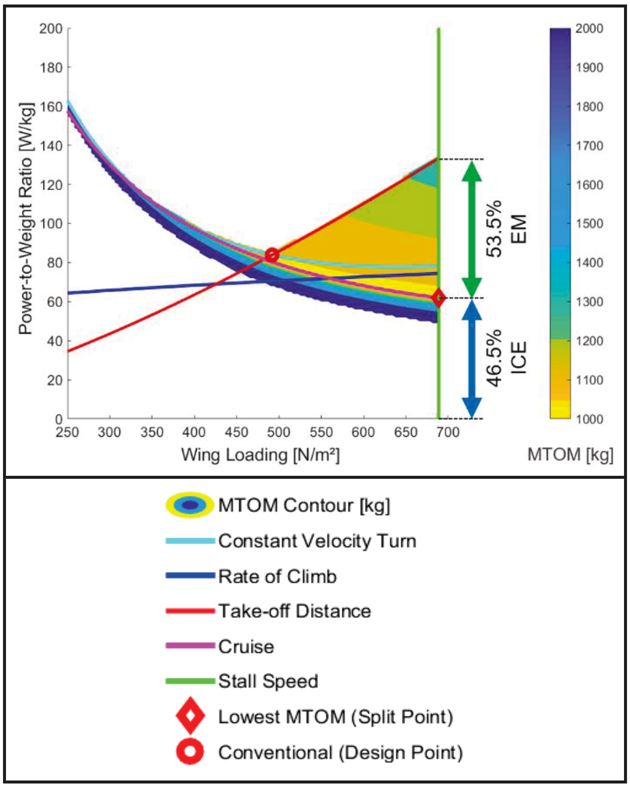

Figure 6 - Design Space of the Baseline Aircraft

Figure 6 reveals that the lightest aircraft are found if the split point is located in the proximity of the cruise constraint curve. The split point of the overall lightest baseline aircraft,

which corresponds to the aircraft with the optimum parallel hybrid design, is placed directly on the cruise constraint.
The design of the ICE to the power required in cruise flight is generally caused by two factors. On the one hand, the use of battery power in cruise flight would lead to an extremely high battery mass and MTOM. This is due to specific energy values of today’s batteries being considerably low in comparison to the values of fossil fuel [20]. On the other hand, the design of an ICE to higher power than demanded in cruise would result in BSFC values during cruise, which do not correspond to the minimum BSFC value of the ICE (according to the used BSFC model). This concludes in a suboptimal fuel consumption in cruise flight and thus, an overall increased fuel mass and MTOM.

The majority of the analyzed designs will feature such a layout of their parallel hybrid propulsion system to achieve a minimal MTOM. However, the amount of mass improvement in comparison to a conventionally powered aircraft and therefore the viability of parallel hybrid aircraft is highly dependent on the characteristics of the flight mission. This correlation is investigated in the following sections.

3.2. Viability Studies on Clean Sheet Designs

Within the scope of the following studies, the viability of clean sheet designs for parallel hybrid GA aircraft is investigated by a comparison to the equivalent conventional design. For that purpose, the most important mission parameters are varied successively during preliminary sizing to receive an overview of the most worthwhile sector for each parameter.
Altogether, five mission parameters are analyzed including the payload, the take-off distance, the cruise velocity, the cruise altitude and the range.

3.2.1. Payload

For this study, the full fuel payload of the baseline aircraft is changed in the range of 150 kg to 450 kg. This conforms to a change in the number of seats of the aircraft from 2 to 6. The payload is not a parameter which has an effect on the constraint diagram of an aircraft. Hence, the design space with its constraints remains unmodified for every regarded value of the payload mass in comparison to the design space of the baseline aircraft (see Figure 6). This implies that also the conventional design point remains at constant position.

Table 3 shows the sizing results of the payload variation. Besides the necessary information for optimal design and split points of each aircraft, the primary energy demand for the optimum design is delivered, as well as the needed fuel mass. Furthermore, for every value of the payload, the respective conventional design point of the sized aircraft is given.

The table illustrates that the position of the overall optimum design point with its split point is also independent from the payload mass, since the values for W/S, P/W and H_P are constant.
However, as expected, the aircraft masses increase with a rising payload mass, just as the used energy. The amount of increase gets lower for higher payload, as the empty mass fraction decreases for higher take-off masses [10].

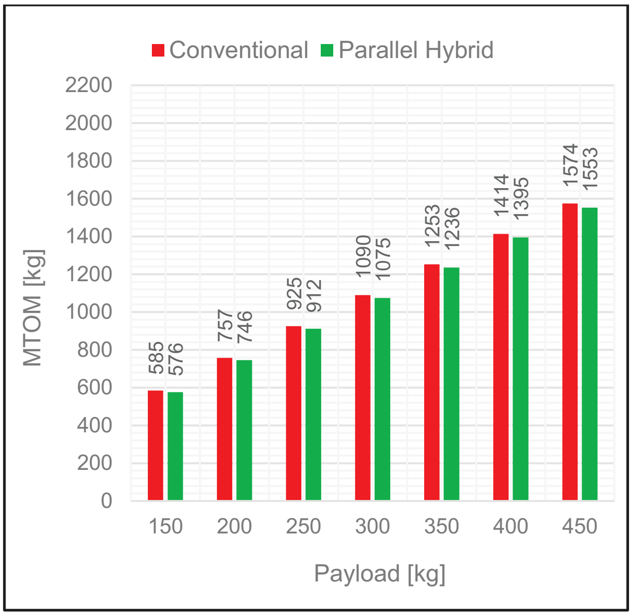

Figure 7 - Comparison for the Payload Variation

Comparing each parallel hybrid solution resulting from the corresponding optimum design point to its conventional

Table 3 - Results of the Payload Variation

Payload [kg]	***150***	***200***	***250***	***300***	***350***	***400***	***450***
MTOM_{opt} [kg]	576	746	912	1075	1236	1395	1553
W/S_{opt} [N/m^2]	689	689	689	689	689	689	689
P/W_{opt} [W/kg]	133	133	133	133	133	133	133
H_P [-]	0.535	0.535	0.535	0.535	0.535	0.535	0.535
$E_{\text{prim,opt}}$ [MJ]	2772	3588	4383	5168	5942	6707	7465
$m_{\text{fuel,opt}}$ [kg]	57.6	74.5	91.1	107.4	123.4	139.3	155.1
$\text{MTOM}_{\text{conv}}$ [kg]	585	757	925	1090	1253	1414	1574
W/S_{conv} [N/m^2]	492	492	492	492	492	492	492
P/W_{conv} [W/kg]	84	84	84	84	84	84	84
$E_{\text{prim,conv}}$ [MJ]	3636	4703	5746	6772	7786	8787	9763
$m_{\text{fuel,conv}}$ [kg]	76.8	99.3	121.4	143.0	164.5	185.6	206.2
ΔMTOM [%]	-1.54	-1.45	-1.41	-1.38	-1.36	-1.34	-1.33
ΔE_{prim} [%]	-23.76	-23.71	-23.72	-23.69	-23.68	-23.67	-23.54

counterpart, it can be observed that for every value of the payload mass, the hybrid solution is always the lighter option (see Figure 7). Yet, the difference in mass is too small to justify a worthwhile design of a parallel hybrid GA aircraft with state of the art technology for the baseline mission, where only the payload is changed. This can be explained by a high system complexity and the associated costs, which outweigh the small advantages in mass.

Once the primary energy use and the fuel mass are consulted as assessment criterions, the viability of parallel hybrid-electric GA aircraft can be rated clearly higher. Both parameters can be improved by nearly 25% for every sweep value of the payload mass. Yet, there is not a value for the payload mass which further improves the viability compared to the parallel hybrid baseline aircraft.

3.2.2. Take-off Distance

Within the scope of this analysis, the take-off distance, respectively the ground run distance, is varied between values of 150 m and 400 m. The take-off distance mainly determines the maximum power that is required in a flight mission and thus dimensions the propulsion unit. The variation of this parameter influences the design space, as it is directly inserted into the take-off distance constraint [10]. Consequently, optimal conventional design points shift towards lower wing loadings for smaller ground run distances.

As expected, Table 4 reveals that the MTOM of all optimal parallel hybrid designs and all optimal conventional designs decreases with increasing ground run distances. However, the extent of the decrease gets lower for higher take-off distances.

The MTOM comparison between the regarded propulsion options for a change of the ground run distance is presented in Figure 8. It exemplifies that parallel hybrid clean sheet designs lead to lighter aircraft for every ground run distance. Again, the differences in mass are very small, especially for medium take-off distances. This lowers the reasonableness of a parallel hybrid-electric layout for light aircraft as described in section 3.2.1. Yet, the trend in the figure shows a significant increase of these differences when the take-off distance is reduced and a slight increase when the distance is extended. This fact can be explained by increasing differences between the required take-off power and the required power in cruise. For conventionally powered aircraft, the ICE has to cover the take-off power and thus, operates at a lower efficiency and a higher fuel consumption in the large cruise segment. For parallel hybrid aircraft, as it is presented in this study, the ICE is designed for the power demanded in cruise, which can be considerably lower if a design at high W/S is envisaged. The EM then only covers the difference in take-off and cruise power. This leads to a higher efficiency and a lower fuel consumption in cruise. Consequently, a major reduction in fuel mass can be accomplished, while the masses of additional motors and batteries are comparatively low.

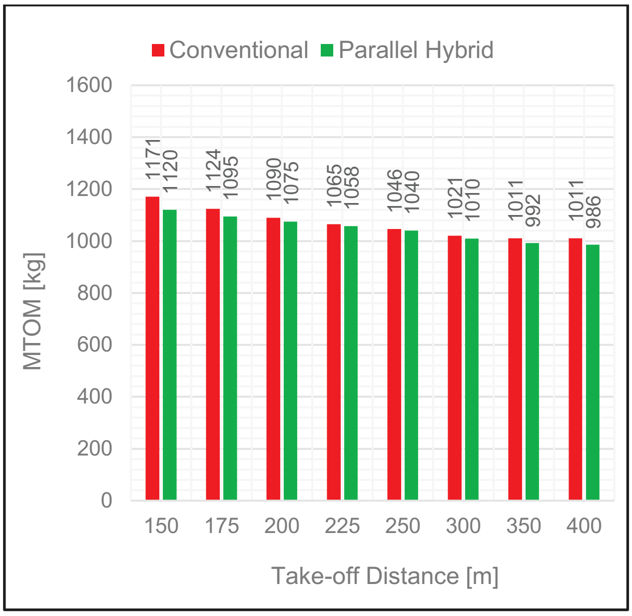

Figure 8 - Comparison for the Take-off Distance Variation

The increased differences between cruise power and take-off power for small ground run distances are achieved by two contributors. On the one hand, the required power for take-off is higher than for larger ground run distances. On the other hand, the design points are located at the highest possible W/S, causing the differences to be bigger than for lower W/S (see Figure 6).

Table 4 - Results of the Take-off Distance Variation

TOD [m]	**150**	**175**	**200**	**225**	**250**	**300**	**350**	**400**
$MTOM_{opt}$ [kg]	1120	1095	1075	1058	1040	1010	992	986
W/S_{opt} [N/m^2]	689	689	689	689	546	602	658	689
P/W_{opt} [W/kg]	173	151	133	120	80	78	78	78
H_P [-]	0.642	0.588	0.535	0.484	0.075	0.130	0.178	0.208
$E_{prim,opt}$ [MJ]	5365	5254	5168	5097	5917	5349	4980	4820
$m_{fuel,opt}$ [kg]	111.2	109.0	107.4	106.0	124.7	112.4	104.5	101.0
$MTOM_{conv}$ [kg]	1171	1124	1090	1065	1046	1021	1011	1011
W/S_{conv} [N/m^2]	435	464	492	519	546	601	639	639
P/W_{conv} [W/kg]	91	87	84	82	80	78	78	78
$E_{prim,conv}$ [MJ]	7956	7270	6772	6403	6117	5707	5522	5522
$m_{fuel,conv}$ [kg]	168.0	153.6	143.0	135.2	129.2	120.6	116.6	116.6
ΔMTOM [%]	-4.36	-2.58	-1.38	-0.66	-0.57	-1.08	-1.88	-2.47
ΔE_{prim} [%]	-32.57	-27.73	-23.69	-20.40	-3.27	-6.27	-9.82	-12.71

For large take-off distances, only the position of the design points at high W/S contributes, leading to lower differences. As can be seen in Table 4, the design points of medium distances shift towards lower W/S. This entails lower degrees of hybridization and smaller differences between take-off power and cruise power.
Accordingly, it can be stated that parallel hybrid-electric designs of GA aircraft are most suitable and viable in terms of mass and cost for missions with extremely low ground run distances. Applications including larger distances do not pay off, when MTOM is the design objective.

The same conclusion can be drawn, if the assessment criterions of primary energy or fuel mass are used. The improvement of both parameters for parallel hybrids decreases from initially 33% at the shortest, to only 3% at a medium take-off distance. A slight recovery up to 13% can be observed for large ground run distances.

3.2.3. Cruise Velocity

For the following study, the velocity in the cruise phase of the baseline mission is increased from a value of 40 m/s to a value of 75 m/s with increments of 5 m/s. The cruise velocity, in combination with the air density, mainly determines the drag force that has to be compensated by the propulsive force in cruise flight. Therefore, it has an influence on the cruise airspeed constraint as well as on the constant velocity turn constraint [10]. This results in a modified design space and a travel of conventional design points.
The results of the cruise velocity study are given in Table 5.

The general trend shows an increase in MTOM of optimal designed parallel hybrid-electric aircraft with higher cruise velocities. The same relation can be observed for the optimal conventional designs. This is due to the increased power demand at higher cruise velocities.

Figure 9 offers the comparison of the MTOM values which have been calculated for the unconventional and the conventional design points corresponding to the different cruise speeds.
It is shown that the difference in mass generally decreases when the cruise velocity is increasing. A higher cruise power is demanded for rising cruise speeds, while the maximum required power remains constant. This leads to a smaller benefit by the design of the ICE to cruise conditions, because the overall power is closer to the required power in cruise. Also, the power that the ICE must deliver for an optimal design exceeds the power that the ICE of the conventionally powered aircraft must deliver. This is due to the fact that the hybridization decreases while the P/W increases (see Table 5), leading higher fuel consumption.
For cruise velocities greater than 70 m/s, the cruise flight defines the overall power requirements. Thus, the design of the ICE to cruise conditions equals to a conventional design with no hybridization. The resulting difference in mass is zero.
Consequently, parallel hybrid-electric GA aircraft featuring a clean sheet design show a low viability measured by their MTOM for high cruise velocities. Yet, the potential of such aircraft including low cruise speeds is significantly higher. Mass reductions in the range of 10% can be expected.

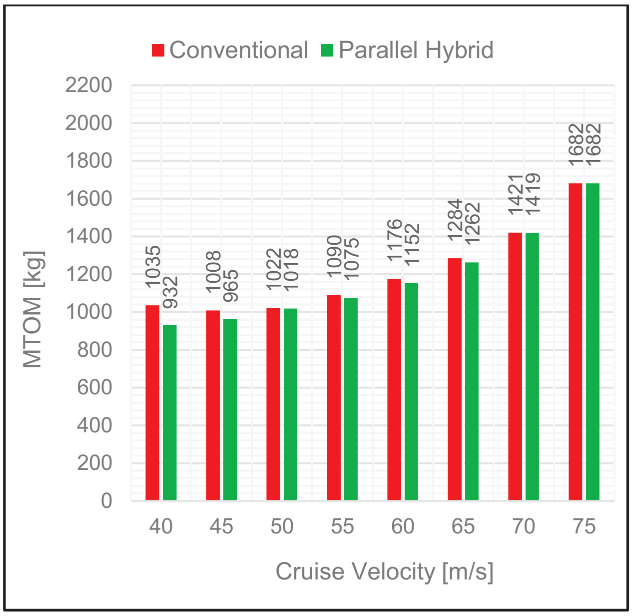

Figure 9 - Comparison for the Cruise Velocity Variation

Table 5 - Results of the Cruise Velocity Variation

v_{cruise} [m/s]	**40**	**45**	**50**	**55**	**60**	**65**	**70**	**75**
$MTOM_{opt}$ [kg]	932	965	1018	1075	1152	1262	1419	1682
W/S_{opt} [N/m^2]	420	426	450	689	689	689	689	689
P/W_{opt} [W/kg]	69	69	74	133	133	133	133	144
H_P [-]	0.390	0.219	0.093	0.535	0.427	0.290	0.123	0.000
$E_{prim,opt}$ [MJ]	4166	4913	5795	5168	6224	7733	9881	13183
$m_{fuel,opt}$ [kg]	86.7	103.0	122.0	107.4	129.9	162.1	208.1	278.4
$MTOM_{conv}$ [kg]	1035	1008	1022	1090	1176	1284	1421	1682
W/S_{conv} [N/m^2]	242	346	444	492	543	597	652	689
P/W_{conv} [W/kg]	64	67	73	84	96	109	124	144
$E_{prim,conv}$ [MJ]	6860	6089	6016	6772	7707	8864	10323	13183
$m_{fuel,conv}$ [kg]	144.9	128.6	127.1	143.0	162.8	187.2	218.0	278.4
$\Delta MTOM$ [%]	-9.95	-4.27	-0.39	-1.38	-2.04	-1.71	-0.14	0.00
ΔE_{prim} [%]	-39.27	-19.31	-3.67	-23.69	-19.24	-12.76	-4.28	0.00

If the primary energy use and the fuel mass are regarded, the viability assessment for this study concludes with the same result. The improvement of these design parameters, in consideration of a clean sheet design, decreases from around 40% at 40 m/s to 0% at 75 m/s.

3.2.4. Altitude

The variation of the altitude incorporates the change of the total height in cruise flight from 1000 m to 4500 m with a step size of 500m. The cruise altitude strongly influences the air density and thus, in interaction with the cruise velocity, the total drag force in the cruise segment. Hence, also the altitude affects the cruise airspeed constraint, the constant velocity turn constraint and the design space itself.

It can be seen that the take-off masses of sized parallel hybrid GA aircraft increase with higher cruise altitudes due to higher power demands (see Table 6). Conventional aircraft can perform superior at higher altitudes, because of the lowered performance of ICEs with decreasing air density. This can increase the power loading and thus the efficiency in cruise flight, as the ICE is designed for higher power demands than those occurring in cruise.
In contrast, optimal designed parallel hybrids cannot benefit from this correlation, since the ICE is already designed for cruise flight. Therefore, higher altitudes solely increase the overall power that the ICE of a parallel hybrid aircraft has to provide, which rises the fuel mass.

The use of parallel hybrid-electric propulsion systems for GA aircraft is highly worthwhile for missions, which include low altitude cruise flights, as illustrated by Figure 10. This results from high differences in MTOM between optimal designs of the conventional option and optimal designs of the parallel hybrid option for small altitudes. For example, a cruise flight at 1000 m with a hybridized aircraft, can reduce the total aircraft mass by 10% in comparison to a conventionally powered aircraft operating at the same altitude.
However, for higher altitudes the viability rapidly decreases, as the mass differences reduce until conventional and hybrid design conform for a value of the altitude of 4000 m. Thereafter, the difference in MTOM and thus the viability rises again, because the conventional aircraft is sized by the power required in cruise for higher values than 4000 m.

When comparing the primary energy usage and the fuel masses required for conventional, respectively parallel hybrid-electric clean sheet designs, the low flight missions perform best in regard to the improvement of these parameters through the hybrid designs, too. This can be substantiated by an enhancement of 34% of both parameters for a sweep value of 1000 m.
The improvements which can be found for higher altitudes, though, are negligible in consideration of the complexity and the costs of hybrid-electric propulsion systems, as well as the development effort for appropriate aircraft utilizing these systems.

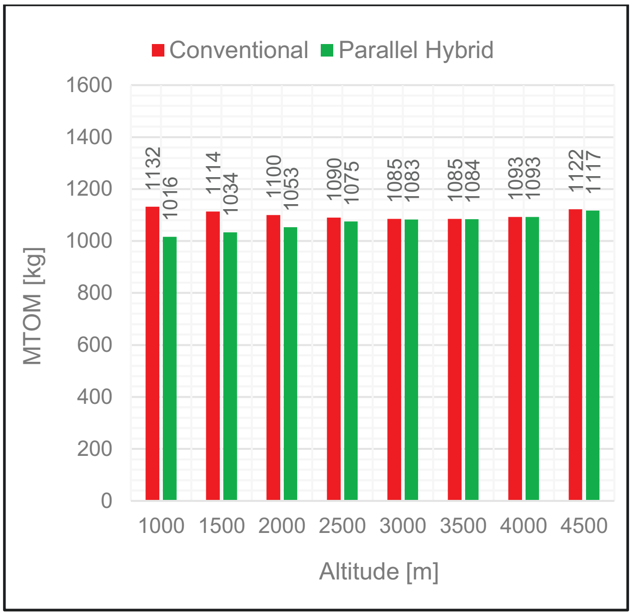

Figure 10 - Comparison for the Altitude Variation

It has to be considered that the results of this comparison are based upon altitude changes, which lead to changes in climb and descent distances.
The range only refers to the distance covered in cruise flight. Therefore, minor changes in energy carrier masses must be taken into account, which are not a direct result of altitude variation.

Table 6 - Results of the Altitude Variation

h_{cruise} [m]	1000	1500	2000	2500	3000	3500	4000	4500
$MTOM_{opt}$ [kg]	1016	1034	1053	1075	1083	1084	1093	1117
W/S_{opt} [N/m^2]	689	689	689	689	502	514	533	562
P/W_{opt} [W/kg]	133	133	133	133	86	89	94	101
H_P [-]	0.566	0.558	0.548	0.535	0.061	0.092	0.000	0.000
$E_{prim,opt}$ [MJ]	5224	5180	5162	5168	6381	6103	6207	6209
$m_{fuel,opt}$ [kg]	109.6	108.3	107.6	107.4	134.5	128.4	131.1	131.2
$MTOM_{conv}$ [kg]	1132	1114	1100	1090	1085	1085	1093	1122
W/S_{conv} [N/m^2]	476	480	485	492	501	514	533	519
P/W_{conv} [W/kg]	80	81	82	84	86	89	94	100
$E_{prim,conv}$ [MJ]	7867	7447	7085	6772	6523	6329	6207	6377
$m_{fuel,conv}$ [kg]	166.2	157.3	149.7	143.0	137.8	133.7	131.1	134.7
ΔMTOM [%]	-10.25	-7.18	-4.27	-1.38	-0.18	-0.09	0.00	-0.45
ΔE_{prim} [%]	-33.60	-30.44	-27.14	-23.69	-2.18	-3.57	0.00	-2.63

3.2.5. Range

To investigate the influence of different design ranges on the viability of parallel hybrid-electric GA aircraft, the range is varied between 500 km and 2000 km, considering increments of 250 km. As mentioned above, the range, as it is defined within the scope of this paper, solely relates to the distance covered in the cruise segment.

An aircraft's range mainly determines the needed energy for a flight mission, as it usually is the longest mission phase. Yet, it does not influence the design space and the included constraints. Therefore, the conventional design point for each corresponding sweep value of this study conforms to the conventional design point of the baseline aircraft.

The sizing results of the range analysis are showcased in Table 7.

The table points out that the take-off masses of both the optimal parallel hybrid designs and the optimal conventional designs grow with a rising cruise range. This is easily understandable, since a larger cruise flight demands more energy, which results in an increasing fuel mass.

However, the amount of increase in fuel mass and MTOM for conventional aircraft is considerably higher than for hybrid aircraft (see Figure 11). Therefore, the viability of parallel hybrid GA aircraft including a clean sheet design increases, when the range is enlarged.

This correlation is, on the one hand, caused by the design of the ICE to cruise conditions, which entails the average BSFC in cruise being close to minimum. For longer cruise distances, an optimal designed hybrid GA aircraft operates at this low BSFC for a longer time. Therefore, it is able to save more and more fuel compared to a conventionally powered aircraft with the same mission requirements, which operates at a higher BSFC because of the design of its propulsion system to the take-off performance.

On the other hand, the optimal design points shift towards higher W/S for larger ranges (see Table 7), resulting in bigger wing areas and thus, higher lift-to-drag ratios in cruise flight. Therefore, besides the propulsive efficiency, also the aerodynamic efficiency is improved by parallel hybrid propulsion systems for increased ranges, lowering the fuel consumption, too.

This implies that there is, theoretically, an open end to this correlation, where the aircraft with the largest ranges can achieve the highest benefits from parallel hybrid designs. However, the MTOM is usually highly dependent on range, such that light aircraft are primarily designed for shorter ranges. Hence, the improvement of MTOM by using parallel hybrid electric propulsion systems for light GA aircraft is restricted to approximately 14%, considering a maximum range of 2000 km.

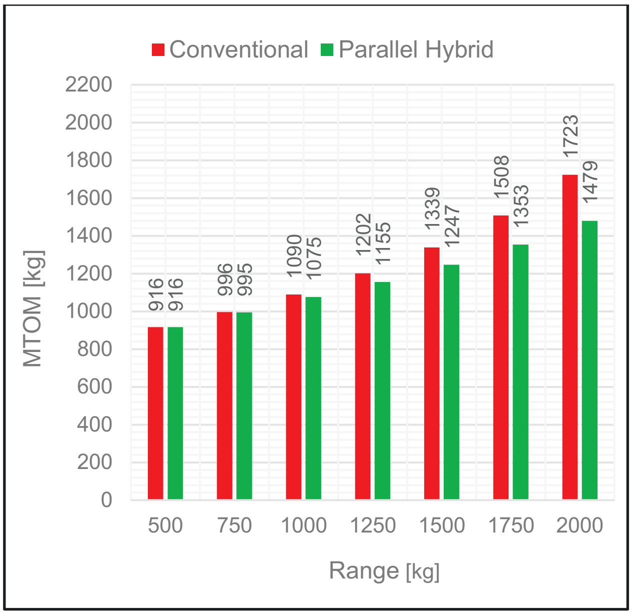

Figure 11 - Comparison for the Range Variation

For the assessment criterions of primary energy use and fuel mass the same results can be observed. A maximum improvement of both parameters by about 35% is possible for long ranges.

Extremely short ranges for parallel hybrid GA aircraft do not offer much benefit over conventionally powered GA aircraft.

Table 7 - Results of the Range Variation

Range [km]	***500***	***750***	***1000***	***1250***	***1500***	***1750***	***2000***
$MTOM_{opt}$ [kg]	916	995	1075	1155	1247	1353	1479
W/S_{opt} [N/m^2]	492	492	689	689	689	689	689
P/W_{opt} [W/kg]	84	84	133	133	133	133	133
H_P [-]	0.040	0.040	0.535	0.535	0.535	0.535	0.535
$E_{prim,opt}$ [MJ]	3177	4784	5168	6727	8530	10625	13099
$m_{fuel,opt}$ [kg]	67.0	100.9	107.4	140.2	178.1	222.2	274.2
$MTOM_{conv}$ [kg]	916	996	1090	1202	1339	1508	1723
W/S_{conv} [N/m^2]	492	492	492	492	492	492	492
P/W_{conv} [W/kg]	84	84	84	84	84	84	84
$E_{prim,conv}$ [MJ]	3209	4840	6772	9101	11947	15499	20033
$m_{fuel,conv}$ [kg]	67.8	102.2	143.0	192.2	252.3	327.4	423.1
$\Delta MTOM$ [%]	0.00	-0.10	-1.38	-3.91	-6.87	-10.28	-14.16
ΔE_{prim} [%]	-1.00	-1.16	-23.69	-26.09	-28.60	-31.45	-34.61

3.3. Viability Studies on Retrofit Designs

Retrofit designs for hybrid-electric powered aircraft might be the simplest alternative to ensure an environmentally friendly aviation in the near future.
In order to retrofit a conventionally powered aircraft into a hybrid-electric powered aircraft, it is necessary to replace the entire propulsion system. While the wing area remains at a constant value, new degrees of freedom arise for the propulsion system design which could lead to more efficient aircraft.
Hereinafter, the viability of retrofit designs with different levels of hybridization is analyzed by a comparison to the conventional baseline aircraft. Within the scope of this study, the mission parameter are not changed. They correspond to those of the baseline mission.

In Table 8, the sizing results of seven possible retrofit aircraft are given as well as the results of the conventional baseline aircraft with a H_P of zero.
The wing areas of all presented aircraft equal approximately 21.7 m^2.

The results indicate that for the reference mission, there is no worthwhile retrofit design with advantages in MTOM in comparison to the baseline.
The lightest possible retrofit offers roughly the same mass as the baseline, while its hybridization is solely 5%. Aside from the unchanged MTOM, the practicability of such a design is low because the low H_P requires a very small EM with an additional integration mass. Still, the ICE size must nearly stay the same.
Designs with higher H_P show a successive increase in MTOM, since the battery use in cruise rises (see Figure 12). A design of the ICE to cruise conditions (see Figure 6) is not possible for any retrofit operating at the baseline mission with the same wing area as the reference aircraft. Thus, the viability of such designs decreases the larger the hybridization gets, when MTOM is the assessment criterion.

In terms of the primary energy use, parallel hybrid retrofit designs with low hybridization offer slight benefits in the range of 1.5%. With respect to the complexity and the cost-intensive installation of a new propulsion system, this benefit does not justify the design effort, too. For higher levels of hybridization this benefit shrinks until the primary energy use increases compared to the baseline. This is due to the rise of the total aircraft mass, which also causes the fuel mass to be almost constant throughout all regarded hybridizations.

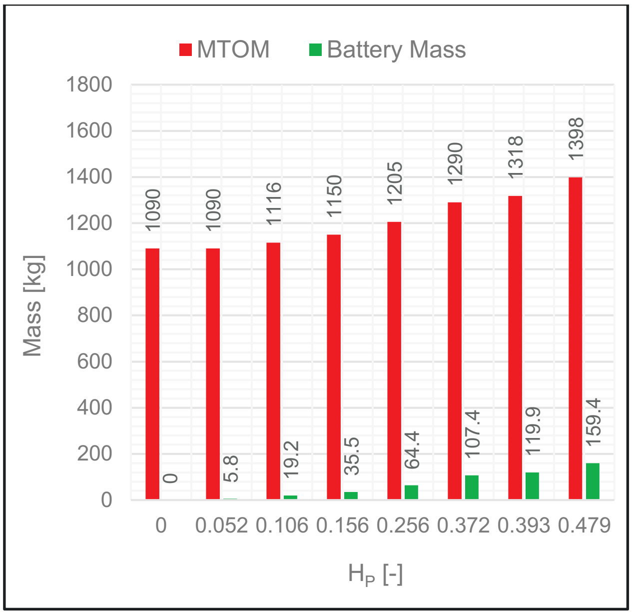

Figure 12 - Comparison for Retrofit Designs

3.4. Advanced Technology Parameter Studies

With steady technological progress, the development of electrified aircraft and their introduction to the market is expected to be simplified. Especially two sectors promise great potential in regard to a major improvement of the feasibility of such aircraft.
On the one hand, the development of batteries towards higher specific energies simplifies electric power utilization in cruise flight. This allows for the use of highly efficient EMs during the whole mission, which offer outstanding energy conversion ratios over a broad power band in comparison to ICEs [21, 22]. The main advantage of an increasing battery technology level, though, is represented by major savings in battery mass.
On the other hand, improved aerodynamic efficiency can positively affect the viability of parallel hybrid aircraft. By decreasing the power demands through drag, the overall

Table 8 - Retrofit Sizing Results

H_P [-]	***0.000***	***0.052***	***0.106***	***0.156***	***0.256***	***0.372***	***0.393***	***0.479***
$MTOM_{opt}$ [kg]	1090	1090	1116	1150	1205	1290	1318	1398
W/S_{opt} [N/m^2]	492	494	506	518	546	586	594	634
P/W_{opt} [W/kg]	84	84	87	90	97	107	109	119
$E_{prim,opt}$ [MJ]	6772	6672	6679	6737	6744	6814	6883	6950
$m_{bat,opt}$ [kg]	0.0	5.8	19.2	35.5	64.4	107.4	119.9	159.4
$m_{fuel,opt}$ [kg]	143.0	140.7	140.2	140.8	139.7	139.3	140.3	140.0
$MTOM_{conv}$ [kg]	1090	1090	1090	1090	1090	1090	1090	1090
W/S_{conv} [N/m^2]	492	492	492	492	492	492	492	492
P/W_{conv} [W/kg]	84	84	84	84	84	84	84	84
$E_{prim,conv}$ [MJ]	6772	6772	6772	6772	6772	6772	6772	6772
$m_{fuel,conv}$ [kg]	143.0	143.0	143.0	143.0	143.0	143.0	143.0	143.0
ΔMTOM [%]	0.00	0.00	2.39	5.50	10.55	18.35	20.92	28.26
ΔE_{prim} [%]	0.00	-1.48	-1.37	-0.52	-0.41	0.62	1.64	2.63

share of installed electric power in the total required power is increased, when the design point stays the same. This results in a greater difference between cruise power and maximum power. While poor BSFC values and high power demands for the ICE are expected for conventional aircraft in consideration of this relation, an optimal designed hybrid-electric aircraft can furtherly profit, as described in section 3.2.2.

In the following sections, sensitivity studies on the battery specific energy and the aerodynamic efficiency, which is represented by the parasitic drag coefficient $c_{D,min}$, are conducted. Their influence on the viability of parallel hybrid GA aircraft is evaluated in comparison to technology's state of the art.

3.4.1. Battery Specific Energy

Statistics indicate that in the past, battery specific energy increased by approximately 7% per year. This trend will most likely not continue due to physicochemical limits of standard electrode materials [20], restricting the specific energy of common Lithium-ion cells to a maximum of 400 Wh/kg [23]. However, research on novel battery technologies is ongoing. E.g. Lithium-sulfur batteries can achieve specific energies in the range of 350 Wh/kg with today's technology, and promise values of 600 Wh/kg for market readiness in 2025 [24].

Based on the mentioned aspects, the battery specific energy assumed for the reference aircraft is increased up to 600 Wh/kg, considering increments of 50 Wh/kg.
The results of the sensitivity study for battery specific energy are illustrated in Table 9.

The table indicates that the MTOM of the optimal designed aircraft decreases with increasing battery specific energy, while the conventional design remains unmodified because the design space is not changed by varying sweep values. The decrease mainly arises from lower battery mass. In comparison to the parallel hybrid baseline aircraft with state of the art technology, the battery mass can be reduced by 61.4% with 600 Wh/kg batteries. Another contributor is the fuel mass, which decreases due to sizing effects that are caused by lowered overall mass and thus lowered power demands.

With respect to the viability of parallel hybrid GA aircraft it can be stated that advanced battery technology is able to further enhance the benefits offered by such propulsion concepts compared to fuel based systems. A maximum benefit of 8.2% improvement in MTOM is achieved within the scope of this study for a sweep value of 600 Wh/kg (see Figure 13).
However, the amount of decrease in MTOM declines with increasing technology level. Therefore, a threshold in battery specific energy exists, after which a continuing rise of the technology level is not worthwhile anymore. Complexity and costs of appropriate battery systems most likely will outweigh the relatively small advantages in mass which can be expected additionally.

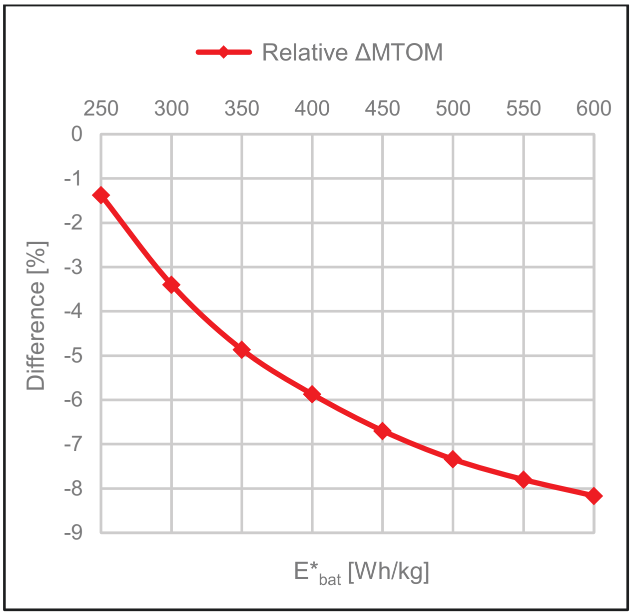

Figure 13 - Battery Specific Energy Sensitivity

Table 9 - Results of the Battery Specific Energy Study

E^*_{bat} [Wh/kg]	***250***	***300***	***350***	***400***	***450***	***500***	***550***	***600***
$MTOM_{opt}$ [kg]	1075	1053	1037	1026	1017	1010	1005	1001
W/S_{opt} [N/m^2]	689	689	689	689	689	689	689	689
P/W_{opt} [W/kg]	133	133	133	133	133	133	133	133
H_P [-]	0.535	0.535	0.535	0.535	0.535	0.535	0.535	0.535
$E_{prim,opt}$ [MJ]	5168	5065	4989	4935	4895	4861	4834	4813
$m_{bat,opt}$ [kg]	40.7	33.3	28.1	24.3	21.5	19.2	17.3	15.7
$m_{fuel,opt}$ [kg]	107.4	105.2	103.6	102.5	101.7	101.0	100.4	100.0
$MTOM_{conv}$ [kg]	1090	1090	1090	1090	1090	1090	1090	1090
W/S_{conv} [N/m^2]	492	492	492	492	492	492	492	492
P/W_{conv} [W/kg]	84	84	84	84	84	84	84	84
$E_{prim,conv}$ [MJ]	6772	6772	6772	6772	6772	6772	6772	6772
$m_{fuel,conv}$ [kg]	143.0	143.0	143.0	143.0	143.0	143.0	143.0	143.0
$\Delta MTOM$ [%]	-1.38	-3.39	-4.86	-5.87	-6.70	-7.34	-7.80	-8.17
ΔE_{prim} [%]	-23.69	-25.21	-26.33	-27.13	-27.72	-28.22	-28.62	-28.93

3.4.2. Aerodynamic Efficiency

In order to ensure efficient and environmentally friendly designs of future aircraft, the improvement of aerodynamic efficiency and therefore the reduction of drag plays an important role. Innovative methods like laminar flow control (LFC) are capable of decreasing the viscous drag of an aircraft up to 60%, which equals to a reduction of absolute aircraft drag of 30% [25].

The evaluation of the effects of these potential drag reductions on the viability of parallel hybrid GA aircraft envisages the improvement of $c_{D,min}$ up to 160 drag counts. This corresponds to a viscous drag reduction of 47% compared to the reference aircraft.

As expected, for decreasing parasitic drag coefficients the comparison indicates a successive reduction in MTOM both for the conventional designs and the optimal parallel hybrid-electric designs (see Figure 14).

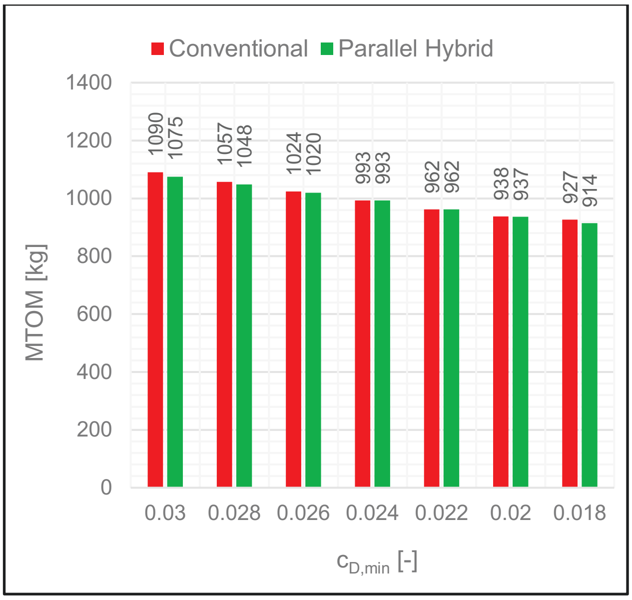

Figure 14 - Comparison for the Aerodynamic Efficiency

As previously mentioned, the main advantage of parasitic drag reduction for parallel hybrid-electric GA aircraft is expected to be caused by decreasing power demands in cruise flight. In this context, the resulting downward shift of the cruise constraint in the design space leads to an inefficient operation of the ICE for the conventional case, while the ICE of the parallel hybrid still is designed for cruise flight.

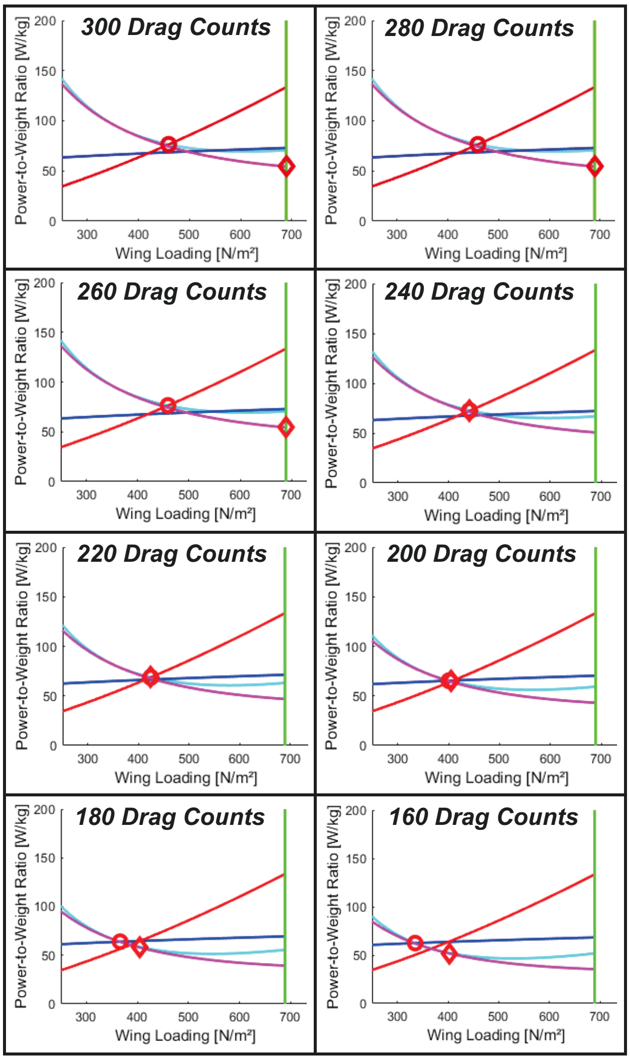

Figure 15 - Design Point Shift

Table 10 - Results of the Aerodynamic Efficiency Study

$c_{D,min}$ [-]	***0.030***	***0.028***	***0.026***	***0.024***	***0.022***	***0.020***	***0.018***	***0.016***
$MTOM_{opt}$ [kg]	1075	1048	1020	993	962	937	914	890
W/S_{opt} [N/m^2]	689	689	689	441	424	405	405	402
P/W_{opt} [W/kg]	133	133	133	73	69	65	65	64
H_P [-]	0.535	0.535	0.535	0.000	0.000	0.000	0.102	0.189
$E_{prim,opt}$ [MJ]	5168	4831	4348	5439	5020	4665	4129	3617
$m_{fuel,opt}$ [kg]	107.4	100.3	90.0	114.9	106.0	98.5	86.9	75.8
$MTOM_{conv}$ [kg]	1090	1057	1024	993	962	938	927	916
W/S_{conv} [N/m^2]	492	475	459	441	424	401	366	335
P/W_{conv} [W/kg]	84	80	76	73	69	65	64	62
$E_{prim,conv}$ [MJ]	6772	6317	5865	5439	5020	4681	4535	4368
$m_{fuel,conv}$ [kg]	143.0	133.4	123.9	114.9	106.0	98.9	95.8	92.2
ΔMTOM [%]	-1.38	-0.85	-0.39	0.00	0.00	-0.11	-1.40	-2.84
ΔE_{prim} [%]	-23.69	-23.52	-25.87	0.00	0.00	-0.34	-8.95	-17.19

This expectation, though, cannot be confirmed by the performed sensitivity study. The mass difference between optimal designed parallel hybrids and the correspondent conventional aircraft does not increase consistently for decreasing parasitic drag. While the improvement in MTOM for the reference aircraft with 300 drag counts amounts to 1.3%, the improvement is zero for values of 240 and 220 drag counts, as both optimal and conventional design points conform. The viability only rises again at a $c_{D,min}$ of 160 drag counts with an enhancement of solely 2.9%. This relative small enhancement is outweigh by additional system weight penalties resulting from innovative drag reduction methods. This lowers the feasibility of parallel hybrid GA aircraft. The inconsistent behavior of the MTOM is due to inconstant positions of overall optimal and optimal conventional design points, caused by the change of constraint functions. Both design points successively move towards lower W/S, which in particular for optimal designed parallel hybrid aircraft leads to an undesired approximation of overall required power and cruise power (see Figure 15).

4. CONCLUSION

In this paper, the viability of parallel hybrid-electric powered GA aircraft with state of the art technology has been assessed for clean sheet designs as well as for retrofit designs. Increasing technology levels have been regarded to investigate the potential of parallel hybrid propulsion for future aviation.

The findings of this paper are based on typical assumptions for GA aircraft. Therefore, the results cannot be generally transferred to all concepts and configurations, which could be realized considering today’s operational and technological opportunities.
Also, the authors are well aware that the included studies were performed with respect to a small range of assessment criteria. Certainly, there are further aspects to evaluate the viability of parallel hybrid GA aircraft.
However, aircraft mass and thus also cost, represent two of the most central aspects to evaluate a new aircraft design.

Throughout all conducted analyses, it was shown that the flight mission has a considerable influence on the feasibility of parallel hybrid GA aircraft, when an optimal design in terms of mass and costs is performed. An ideal mission profile was identified which includes a slow and long-range cruise flight at a low altitude. Additionally, small take-off distances should be considered, which are desirable for short take-off and landing (STOL) applications. With respect to this flight mission, primarily the increase of the range is capable of rising the viability. While small cruise velocities and low altitudes promise large improvements compared to the conventional counterparts too, short take-off distances and high payload do not offer extensive enhancements.
For clean sheet designs, such a mission profile leads to major advantages of parallel hybrid-electric aircraft in comparison to conventionally powered aircraft. Therefore, it can be stated that parallel hybrid GA aircraft featuring a clean sheet design are certainly viable, although the costs and the complexity of their propulsion systems must be considered.
In contrast, it was observed that retrofitted parallel hybrid-electric GA aircraft with today’s technology do not offer worthwhile designs in regard to mass and costs, when respecting the significant installation effort. Whereas for low degrees of hybridization the mass roughly corresponds to the mass of a conventional aircraft, for higher levels of hybridization it steadily increases.

The analysis of increasing technology levels was conducted for battery specific energy and aerodynamic efficiency. The sensitivity study for advanced battery technology showed great potential for the feasibility of parallel hybrid GA aircraft. It predicted potential mass improvements of 8.2% in comparison to the appropriate conventional aircraft with the same mission profile. This improvement could even be enlarged by applying the optimal flight mission presented above.
The perspectives of increased aerodynamic efficiency for the viability of future parallel hybrid aircraft designs, however, appear to be not as promising. For the conducted study, small reductions in aerodynamic drag concluded in an approximation of conventional and overall optimal designs and therefore in a reduction of viability. Solely large reductions in drag showed a slight recovery of the practicability of parallel hybrid designs in terms of MTOM, which is overcome by high technological effort for heavy drag reduction systems.

This work indicates that parallel hybrid-electric GA aircraft can be highly superior to their conventionally powered competitors already. Yet, appropriate design conditions must be taken into account to make full use of the benefits, such that the drawbacks that come along with the complex propulsion system are outweighed.

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