Sizing Considerations of an Electric Ducted Fan for Hybrid Energy Aircraft

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ABSTRACT

Hybrid and universally-electric aircraft are promising configurations to potentially handle the ambitious emission reduction targets set for example by the Advisory Council of Aviation Research in Europe with the Strategic Research and Innovation Agenda. Those concepts are confronted usually with high mass impacts caused by the electric system architecture design. In this paper a closer look is taken on the sizing of the electric power train components and sizing options of the electric motor powering a ducted fan for a discrete parallel hybrid-electric aircraft concept. In such a configuration, the electric system is used as an assistance system to enable a downsizing of the gas turbine and to increase the overall power train efficiency. The studies include considerations concerning geared versus direct drive systems and its implications on overall aircraft level. Furthermore, using the electric system as assistance system different motorization options are discussed to identify the sizing point offering the best utilization of the electric system during the entire mission. As baseline an A320 class aircraft is used for an entry into service year of 2035+ and a 1300nm design range. Finally, for this aircraft concept trade-factors will be provided to cover the impact of a changed electric system mass and efficiency on the overall aircraft mass.

KEYWORDS: electric ducted fan, electric architecture, hybrid-electric aircraft, thrust envelope

NOMENCLATURE

ACARE – Advisory Council of Aviation Research in Europe
ATLeR – Aircraft Top Level Requirements
b – Wing Span
BCU – Battery Control Unit
EDF – Electric Ducted Fan
EPS – Energy and Power Supply
GTF – Geared Turbofan
H_E – Degree of Energy Hybridization
H_P – Degree of Power Hybridization
HTS – High-Temperature Superconducting
MTOW – Maximum Take-Off Weight
PMAD – Power Management and Distribution
OEI – One-Engine Inoperative
OMI – One-Motor Inoperative
SRIA – Strategic Research and Innovation Agenda
SOC – State-of-Charge
SSPC – Solid State Power Controller
TOC – Top-of-Climb
TSFC – Thrust Specific Fuel Consumption
TSPC – Thrust Specific Power Consumption
UEA – Universally-Electric Aircraft

Subscripts
Bat – Battery
Elec – Electric Power Chain
Fuel – Fuel / Kerosene
GT – Gas Turbine
Mot – Motor
N – Net Thrust

1 INTRODUCTION

Electro-mobility in the aeronautical sector seems to be one potential solution to cope with the ambitious emission reduction goals unveiled by the Advisory Council of Aviation Research in Europe (ACARE) [1] with the Strategic Research and Innovation Agenda (SRIA). Those targets specify a reduction of 60% carbon dioxide, 84% nitrogen oxides and 55% perceived noise level based on a year 2000 datum for future transport aircraft with an entry-into-service of 2035. A main focus to fulfil those targets can be performed with an optimization of the energy and power system (EPS). A promising solution seems to be an electric transmission system, which allows for a decoupling of the power sources and the power consumers. For fixed wing aircraft several studies were already performed to cover such new transmission systems for example with turboelectric concepts performed by the NASA N+3X concept [2], even universally-electric aircraft (UEA) like the Bauhaus Luftfahrt Ce-Liner [3] or the Airbus VoltAir [4] and also hybrid-electric aircraft like the Boeing SUGAR Volt [5] with the different topology options like serial, parallel or serial-parallel. The previous studies have already identified that short to medium range aircraft applications utilizing a hybrid-electric EPS enabling block fuel saving potentials of up to 20% [6]. A major challenge, which all different kinds of electric propulsion systems for transport aircraft have currently in common, is the high system mass impact of the new power train. One mass driver is caused by the take-off sizing case of the electric system. The air-density dependent propulsors (ducted fans or propellers) normally show a power lapse over the flight envelope, which an electric motor system does not represent. For example a ducted fan requires only a third of the take-off power during cruise mainly caused by the aircraft top level requirements (ATLeRs) of a typical transport aircraft as presented by [7]. This means that the electric system is sized for a power demand, which is only required for a short time during the mission and implies that this system is running in relative deep part load during cruise. Several design studies identifying an optimal design point of electrically powered propellers and ducted fans have been investigated by [7] and [8]. The focus of those papers was to identify optimal propulsor diameters and tip speeds for an efficient and mass optimized propulsor system. The electric system components were considered as black boxes with constant efficiencies. The included electric motor performance limit was only covered by the maximum installed power.

The main scope of the present paper is to deepen the understanding of electric component designs when linking them with a ducted fan for different sizing cases, using more detailed models. First a brief overview of current concepts of electric propulsion systems including important assessment parameters will be given, followed by the modelling approach of the involved components and subsystems. In the next step different motorization options are presented on a system level without aircraft impact (uninstalled performance). After the identification of the best system parameters for an electric powered ducted fan concept the sensitivities of the system are assessed on overall aircraft level with the help of trade factors. The aim of this study is to identify the impact on overall aircraft level when trading overall system mass versus system efficiency.

2 OVERVIEW OF ELECTRIC PROPULSION TRANSMISSION SYSTEMS OF AIRCRAFT

A big advantage of electric power transmission for propulsion systems is the possibility to mechanically decouple the energy sources like gas turbines or batteries from the power consumers like propulsors. This decoupling leads to a more flexible way to integrate the propulsion system with the airframe and offers new degrees of freedom also concerning options for different energy sources. In case of dual-energy aircraft the most common one is the combination of a conventional kerosene powered gas turbine with a battery. Based on these combinations three different topologies, serial, parallel and serial-parallel, may be considered.

In case of the serial topology the main power transmission is performed electrically, which means that a gas turbine usually converts its generated power with the help of a generator into electrical energy, which is assisted by a battery [9]. This layout allows for a complete decoupling of the propulsors from the gas turbine, which is offering an optimized operation of the gas turbine with regard to efficiency. A drawback of this configuration is that the electric power train has to be sized for the entire propulsive power. This drawback can be avoided by the parallel hybrid topology, where the electric power train is only sized for part of the required propulsive power. Because the main transmission mode is performed mechanically in this case, the gas turbine is still linked to the propulsors, and cannot offer the same flexibility in operation like the serial hybrid. The third type is represented by the serial-parallel topology combining the philosophies of both previously described topologies. The ability of this option is the flexible change between the parallel and the series topology during operation. However, this flexibility causes also the highest complexity with regard to involved components and the associated control system. For that reason this topology type is more of interest for ground based vehicles and not the preferred solution for airborne applications.

A dual-energy aircraft can be fully described independently of the underlying topology by the use of two descriptors proposed by [9], the degree of power hybridization, H_P, given in Eq. 1

$$H_P = \frac{P_{Mot}}{P_{Mot} + P_{GT}} \tag{1}$$

and the degree of energy hybridization, H_E, stated in Eq. 2

$$H_E = \frac{E_{Elec}}{E_{Elec} + E_{Fuel}} \tag{2}$$

H_P represents the ratio of the electric motor shaft power, P_{Mot}, directly powering a thrust producing device to the total shaft power, while H_E covers the ratio of the consumed electric energy, E_{Elec}, for example in form of batteries to the total consumed energy including the fuel based energy, E_{Fuel}. Examples of different aircraft concepts for various combinations of H_P and H_E are visualized in the degree of power and energy hybridization diagram in Fig. 1. Three corner points represent the single energy source extrema with the conventional (H_P=0, H_E=0), the fully turbo-electric (H_P=1, H_E=0) and the universally-electric (H_P=1, H_E=1) aircraft. In between different combinations of series and parallel hybrid topologies cover dual-energy options, in this case different combinations of batteries and kerosene based systems. All concepts within this design space have in common that they have to handle electric power transmission including their new characteristics when combining an air density dependent thrust producing device in combination with an electric system architecture. From an overall systems point of view the partially series hybrid and the parallel hybrid topologies offer the highest degree of freedom with regard to the kind of power supply and the power transfer itself to the thrust producing devices.

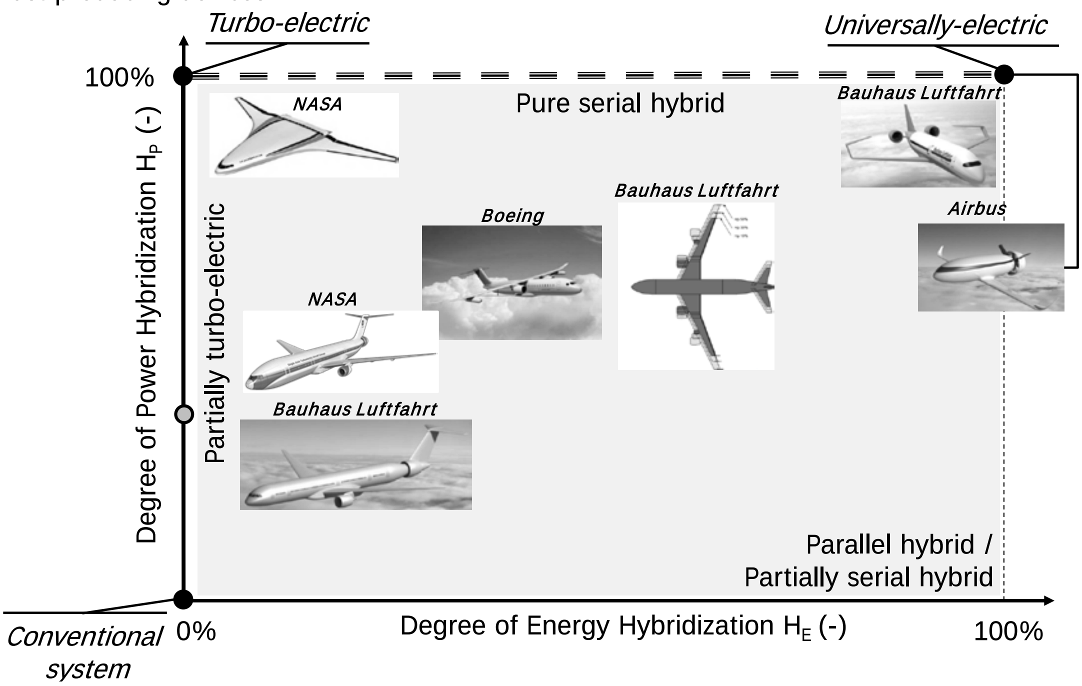

Figure 1: Various turbo-, hybrid- and universally electric concepts marked in the degree of power and energy hybridization diagram for battery-kerosene hybrid-electric aircraft

For the assessment of the performance of hybrid-electric power trains the thrust specific power consumption (TSPC) may be a valid parameter as stated in Eq. 3, which was introduced by [7] and is an enhancement of the thrust specific fuel consumption (TSFC).

$$TSPC = \frac{P_{Supply}}{F_N} = \frac{P_{Bat} + P_{Fuel}}{F_N} \tag{3}$$

The TSPC is defined as the ratio of the supplied power, P_{Supply}, to the system in form of effective fuel enthalpy power, P_{Fuel}, and the electric supplied power of the battery, P_{Bat}, to the generated net thrust, F_N.

3 REFERENCE AIRCRAFT PLATFORMS

For the studies conducted in this paper the reference platform of [10] is used as sketched in Fig. 2, which represents a discrete parallel hybrid-electric aircraft. This configuration features in total four thrust producing devices divided into two conventional gas turbines placed at the inner side of the wing and two fully electric ducted fans (EDF) powered by a battery supplied electrical systems architecture. Those two different propulsion systems are not mechanically or electrically coupled, which means that they can be controlled fully individually, and, therefore, can be categorized according to [11] as a discrete parallel hybrid-electric topology. This platform is sized for a 1300nm mission accommodating 180 PAX. An optimal H_P was identified for this configuration with 30% [10]. The hybridization strategy in this case was to assist the gas turbine during take-off, climb, cruise and diversion. In the other flight phases as well as during reserve mission points like hold only the gas turbines alone are powering the aircraft. According to [10] in this initial study no additional drag due to wind-milling effects was taken into account.

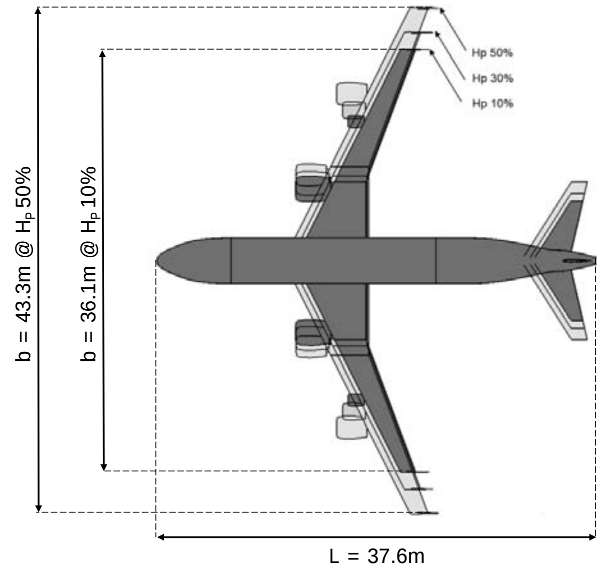

Figure 2: Top view of the year 2035 the discrete parallel hybrid-electric derivative for different degrees of power hybridization [10]

The reference values of this platform are summarized in Table 1 with regard to the hybrid power train mass, the maximum take-off weight (MTOW) and the thrust requirements for top-of-climb (TOC) and one-engine inoperative (OEI) take-off.

Table 1: Reference values and thrust requirements of the A320 based reference aircraft and the discrete parallel hybrid-electric aircraft for a H_P of 30% [10]

Reference Parameter	Conventional A320 based Aircraft	Hybrid-Electric Aircraft
Total Propulsion Mass	4132 kg	24011 kg
GTFs and EDFs Mass	4132 kg	5117 kg
PMAD	n/a	1348 kg
MTOW	60840 kg	77730 kg
TOC Thrust	18.1 kN	46.0 kN
OEI Thrust*	84.0 kN	107.6 kN
* calculated with presented methods for TOC thrust requirement		

4 HYBRID-ELECTRIC POWER TRAIN METHODS

Fig. 3 contains typical components which are involved in an electrically powered ducted fan system. The following section describes the modelling approach for the design and off-design performance of the electric power systems architecture, the gear box system as well as the ducted fan for different thrust requirements.

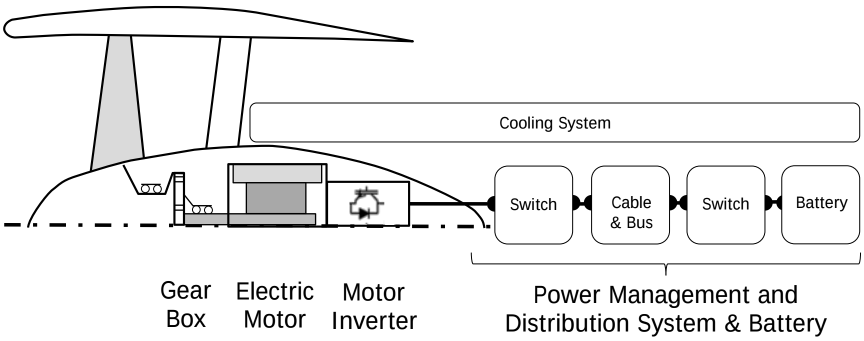

Figure 3: Typical components required for an electrically powered ducted fan

4.1 Electric Systems Architecture Model

An electric system architecture consists of the three main parts energy and power supply (EPS), power management and distribution (PMAD) system and power consumers as outlined in Fig. 4. As EPS batteries are used, which are supplying electric motors with the required power in form of a certain voltage and electric current. The electric power is transmitted with the help of a PMAD system, which is responsible for a safe and redundant transfer to the loads. The battery system normally shows a strongly non-linear discharge behaviour, where the output voltage depends on the required power demand and the current state-of-charge (SOC) as investigated by [12]. Lithium based battery systems currently seem to be a good solution for the energy supply of such architectures. For the transmission of the power there are also several options possible ranging from stabilizing the voltage drop of the battery to a constant transmission voltage or allowing a variable voltage transmission over the PMAD. Based on the results of [13] the most efficient way is the variable system voltage, which is also taken into account for the present studies.

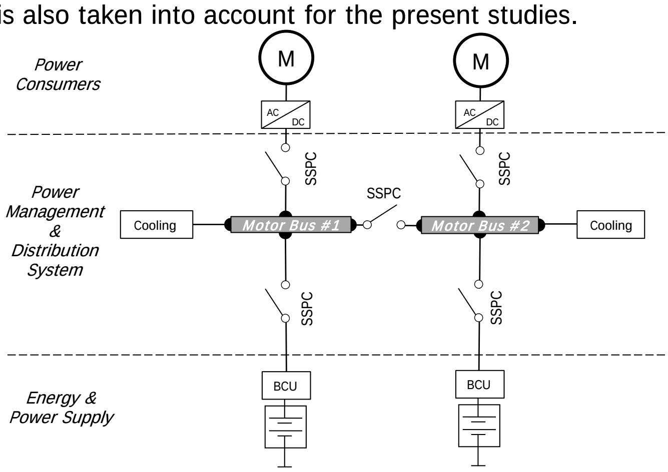

Figure 4: Main components of a battery supplied electric systems architecture design with two separated propulsion chains in normal operation

For the single involved components different models based on handbook methods and scaling relations are used to cover masses and efficiencies also in off-design conditions [12]–[15]. This includes also the impact of different technologies and sensitivities with regard to voltage, transmitted power and switching frequencies. The efficiency and mass ranges covered by those single components are summarized in Table 2.

Table 2: Overview of component model properties

Component	Efficiency Range	Specific Mass Range
Electric Motor	~99.5%	7.0 – 25.0 kW/kg
Inverter/Controller	98.0% – 99.5%	10.0 – 20.0 kW/kg
Aluminum Cables	99.5% – 100%	1.0 – 20.0 kg/m
Protection Switches	99.9%	20.0 – 30.0 kW/kg
Liquid Cooling System [16]	n/a	1.2 kW/kg
Cryocooler [17]	30.0%	0.3 kW/kg
Battery	96.0%-99.5%	1000 Wh/kg*
* assumed technology target for future battery systems		

A special focus in this study will be laid on the electric motor design and the matching of the typical performance map of an electric motor with the ducted fan. The performance characteristic of an electric motor can be characterized in a torque-rotational speed diagram, which is generically sketched in Fig. 5. This chart can be divided into two regions. The first region is the constant torque region, which is normally limited by the maximum torque or electric current, respectively. The second region is the constant power or flux weakening region, which is limited by the maximum electric motor power, in the first instance. In the transition area of both regions normally the design point of the electric motor is defined as symbolized in Fig. 5. In this sketch also typical operating points with regard to rotational speed and torque are marked for TOC and cruise conditions, when sizing the electric motor system for take-off power. The efficiency of an electric motor depends on the rotational speed and torque, which can be translated into electric current and voltage parameters, and is indicated as contours in the figure below.

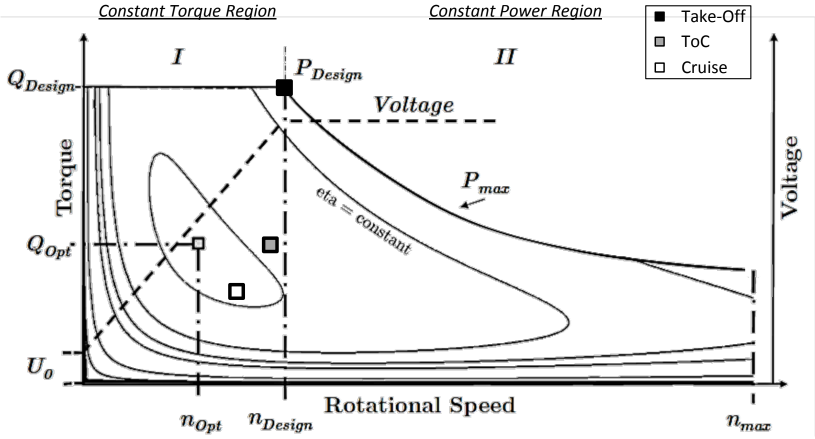

Figure 5: Generic performance chart of an electric motor based on [14]

For the electric motors different types and technologies are available like asynchronous or synchronous machines, or radial or axial flux motors [15]. For the purpose of this paper the focus will be put on synchronous machines based on permanent magnet motors, which seem to be an eligible motor type for an aviation application with regard to mass and efficiency [14]. For high power requirements in the range of megawatts high-temperature superconducting (HTS) motors seem to be a promising solution, even when considering the required cryogenic cooling equipment [14]. This motor type is considered for further studies within this paper.

Full HTS motors, which are using HTS material on the rotor as well as the stator side, can theoretically reach specific powers of up to 40kW/kg [18]. For the modelling approach conducted in this study the methods described in [14] are used, which are covering sensitivities with regard to design shaft power and chosen rotational speed. With this approach specific powers of up to 25kW/kg can be demonstrated, which is in good agreement with numbers published by [17]. Increasing the electric motor speed by 50%, which in turn leads to a torque reduction of about 67%, reduces the electric motor mass by about 40% (see Fig. 6).

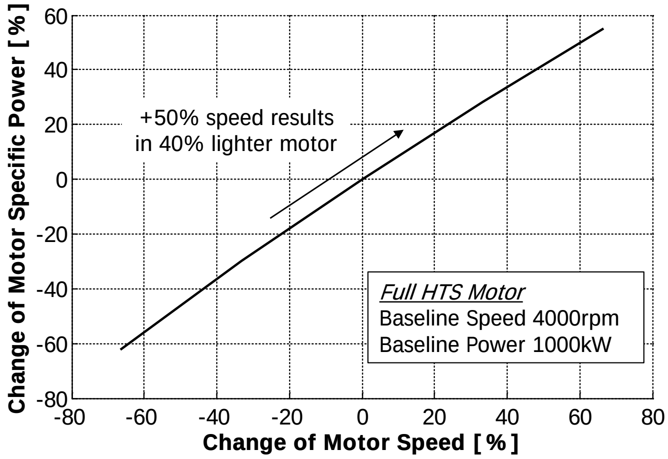

Figure 6: Impact of the rotational speed on the electric motor mass (without cooling and inverter)

The second important component beside the electric motor is the battery. The battery in this study represents an advanced Lithium based cell chemistry and the performance of the battery is modelled according to methods described in [12]. The battery performance is also influencing the other components within the electric power train chain due to voltage variation during the discharge process and the actual state-of-charge (SOC), which is covered by the applied method.

4.2 Ducted Fan Model

The ducted fan model is based on methods described in [19]. This method uses, in the first instance, a 0-dimensional thermodynamic performance model and the standard compressor theory to predict the available thrust as well as the propulsive efficiency of the fan for different flight altitudes, Mach numbers and thrust lever positions. The flow path sizing is conducted in TOC conditions. Based on the propulsion power, which is defined as net thrust times the flight speed, the required fan shaft power can be determined using the propulsive efficiency and the transmission efficiency. The transmission efficiency includes the fan polytropic efficiency, intake, ducting as well as nozzle losses [20].

As mentioned in the previous section of the electric system architecture design, the performance of an electric motor can be described in a torque-rotational speed chart, which is independent of the flight altitude and speed. For the identification of optimal sizing points of the fan characteristic with the electric motor, the standard compressor map, which is a function of the reduced mass flow and the pressure ratio, has to be transferred to such a form. This approach is conducted by varying the thrust lever position for different altitudes and Mach numbers to cover the full fan envelope, as generically visualized in Fig. 7.

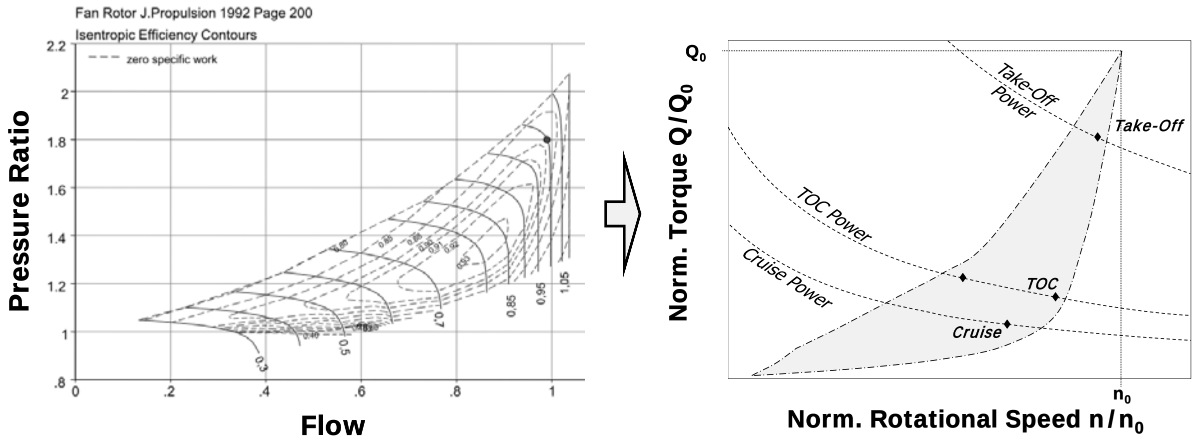

Figure 7: Conventional fan map as function of pressure ratio and mass flow (left) [21]; transformed generic fan map as a function of rotational speed and shaft torque (right)

4.3 Gear Box Model

A gear box system may be required between the fan shaft and the electric motor system. As mentioned in Section 4.1 the mass of the electric motor decreases when increasing the rotational speed. Because the fan speed is limited by aerodynamics, the electric motor speed can be used as additional degree of freedom by implementing a gear box between the fan and the motor shaft. For this purpose a planetary reduction gear box system is considered and the mass, m_{Gear}, is modelled in the sizing case according to methods described in [22] including the reference values with Eq. 4

$$m_{Gear} = m_{Ref} \cdot k \cdot \left(\frac{GR}{GR_{Ref}} \right)^{0.15} \cdot \frac{Q_0}{Q_{Ref}} \tag{4}$$

$$with \quad GR = \frac{n_{Mot}}{n_{Fan}}$$

The gear box mass is scaled with the current gear ratio, GR, which is the ratio of the motor speed, n_{Mot}, to the fan speed, n_{Fan}, a reference gear ratio, GR_{Ref}, the design torque of the new gear box system, Q_0, and a reference torque, Q_{Ref}. A design efficiency of 99.5% is assumed.

For the estimation of the off-design efficiency of the gear box system a normalized performance chart is used, see Fig. 7 derived from [23]. The normalization is based on the design values, which are used to estimate the gear box mass in Eq. 4.

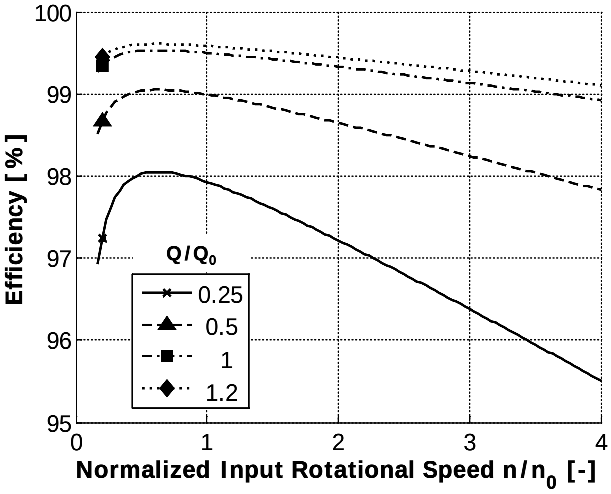

Figure 8: Normalized off-design performance of a single stage planetary gear box based on [23]

4.4 Gas Turbine Characteristics

Beside the electric power train, also the conventional gas turbine is an important contributor in a parallel hybrid-electric power train. The conventional gas turbine model is taken from [6]. The architecture is based on a geared turbofan (GTF), which performance characteristics are simulated with the engine performance tool GasTurb2011® [24]. The GTF is featuring a gear box with a GR of 3 and is sized to TOC conditions at 35,000ft at Mach 0.78. The model covers typical design laws to consider component efficiencies, cooling and size impacts according to [25] and [20]. Those laws, mainly for the component efficiencies, have been adapted to project the technology and in turn performance for an EIS year of 2035. With this adaption the TSFC varies between 13.24 g/kN/s and 13.30 g/kN/s depending on the GTF size in the design point. For the estimation of the mass of the GTF, a standard approach is used based on the geometry of the individual components and the performance parameters. According to [6] this approach leads to a specific thrust for a 17.7 kN GTF of 8.77 N/kg.

5 SYSTEM SIZING AND PERFORMANCE STUDIES

Based on the hybrid-electric reference platform presented in Section 3 different options for the motorization are conducted to identify the optimal utilization of an electric system. For this purpose all conducted studies are performed in a first step on system level, where the EDF is sized to the reference thrust of the discrete parallel hybrid-electric aircraft.

5.1 Electric-Ducted Fan Motor Coupling Options

In the first study the different mechanical coupling options are discussed, if a direct drive system offers a better overall system performance compared to a geared system. The disadvantage of a geared system is the additional required component within the power train, which causes additional weight and efficiency losses. Nevertheless, a reduction gear box system has the potential to significantly reduce the motor system weight as indicated in Section 4.1. In the first assessment only the direct influenced components are considered, which are the ducted fan, the gear box and the electric motor including the corresponding inverter-controller unit and the cooling system. The electric motor is in this case scaled to meet the new rotational speed requirements by not changing the motor architecture like number of pole pairs. The results of this approach for different gear ratios and design thrusts are given in Fig. 9. It is obvious that with increasing gear box ratio the efficiency is slightly decreased compared to the direct drive system. From a mass point of view saving potentials of up to 35% are possible. At this stage, the impact of the system efficiency and mass on overall aircraft level is not known, assuming for further studies a GR of 2, showing in the first instance a good trade-off between mass and efficiency reduction.

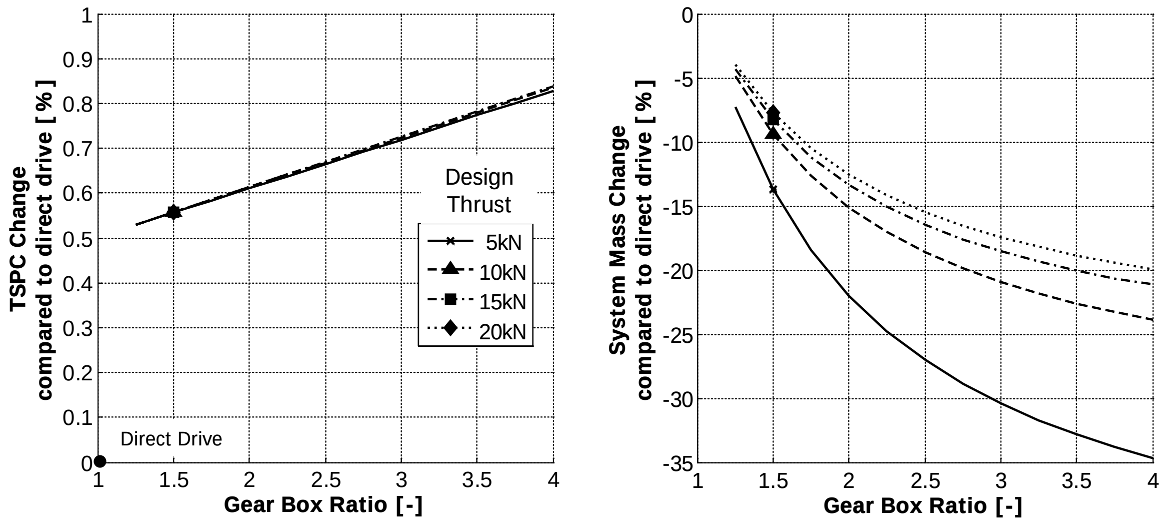

Figure 9: Impact of the design gear ratio on the system efficiency (left) and mass (right)

The second aspect, which has to be discussed, is the identification of the potential sizing points of the electric motor system. For that purpose the standard compressor maps have to be transferred to a torque rotational speed diagram as indicated in Section 4.2. Fig. 10 shows the results of such an approach, which is valid for a design thrust range between 5kN and 25kN using the described methods. The torque demand between the TOC sizing point and the take-off point is 3 times higher and at an approximately higher rotational speed of 8%.

Therefore, for a discrete parallel hybrid-electric power train configuration, where the electric system is used as assistance, three different design points of the electric motor can be considered, which are

1. The motorization for the take-off power demand
2. The sizing of the electric motor for the TOC power demand
3. The motorization for the TOC power demand, but at corresponding take-off rotational speed

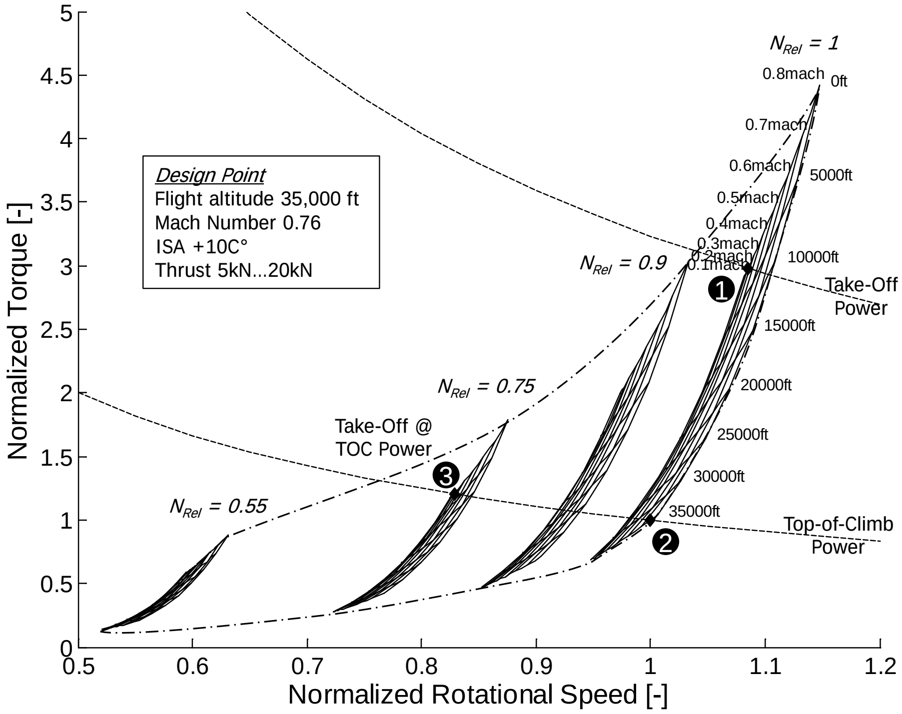

Figure 10: Normalized rotational speed and torque diagram of the ducted fan model at different thrust lever positions (normalized to TOC conditions)

5.2 Performance of Different Motorization Options

For the different motorization options, which were identified in the previous section, the design performance for the reference thrust of 6.8 kN as defined in Section 3 as well as the resulting take-off thrust are investigated. For scenario Sizing No. 1 the electric motor and the gear box are sized for the take-off power and rotational speed demands of the ducted fan, which is sized for the TOC thrust of 6.8 kN. In scenario Sizing No. 2 the electric motor and the gear box system are designed for the TOC conditions. In the case of Sizing No. 3 the electric motor and the gear box system are sized for the TOC power, but at take-off shaft rotational speed of the ducted fan, which is rated during this flight condition to meet the TOC power demand. The results of this approach are summarized in Table 3 concerning mass and efficiencies of the different options.

Table 3: Results of different motorization options for an identical sized ducted fan

	Sizing No.1		Sizing No.2		Sizing No.3	
	Take-Off	TOC	Take-Off	TOC	Take-Off	TOC
Thrust [kN]	37.1	6.8	12.6	6.8	15.6	6.8
Motor Shaft Power [kW]	6777	2116	1625	2095	2095	2095
Total Efficiency [%]	36.7	70.6	52.0	72.4	49.9	72.2
Electric Motor [%]	99.1	98.7	99.1	99.1	99.1	99.1
PMAD [%]	97.8	96.6	97.9	97.7	97.8	97.5
Gear Box [%]	99.5	98.5	99.5	99.5	99.5	99.4
Fan [%]	38.1	75.2	53.9	75.2	51.7	75.2
Total Mass [kg]	2152		998		1021	
Electric Motor [kg]	270		104		122	
PMAD [kg]	1400		447		447	
Gear Box [kg]	51		17		21	
Fan [kg]	430		430		430	
Specific Power [kW/kg]	3.15		1.63		2.05	
Specific Thrust [N/kg]	17.24		12.63		15.28	

Sizing No. 1 causes the highest system mass of 2152 kg with a specific power of 3.15 kW/kg and an overall TOC efficiency of 70.6%. In case of Sizing No.2 the take-off thrust is reduced by 66.0% compared to Sizing No.1, but the EDF masses have also been reduced by about 53.6%. The TOC overall EDF efficiency increases to 72.4%. By changing the design point of the electric motor to meet the TOC equivalent power at take-off as performed with Sizing No. 3 the take-off thrust can be increased by 23.6% compared to Sizing No. 2, which is still 58.0% less than for Sizing No. 1. Nevertheless, due to this change the electric motor is getting 17.3% and the gear box 23.5% heavier compared to Sizing No. 2, but the overall EDF system mass only increases by 2.3%. The electric motor has been slightly oversized compared to Sizing No. 2, because due to the changed sizing point of the gear box system to take-off conditions the gear box efficiency is not equal to Sizing No. 2. To meet again the TOC power demand, this efficiency drop has to be compensated by a higher torque and in turn power demand. However, considering the electric power train only as an assistance system, this design option seems to be a good utilization of the EDF system. Using the electric system as standalone thrust producing device Sizing No. 1 seems to be the only possible solution, if the ATLeRs should be fulfilled and the EDF design is not adapted.

Fig. 11 indicates for the different motorization options the uninstalled (without aircraft) thrust limiting parameters as well as the maximum available thrust in different flight states. From this figure it can be seen that in case of Sizing No. 1 only the fan is the thrust limiting component. The electric motor is constraining the maximum thrust by the torque and power in altitudes below 10,000ft at high Mach numbers, which is normally outside of the flight envelope. In total this option leads to a thrust ratio, which is defined as a maximum thrust to TOC thrust, of up to 6. In case of Sizing No. 2 the electric motor is almost limiting the thrust by the torque in all flight phases. Only at high altitudes at low Mach numbers the fan is the constraining component, which is normally not reached due to the stall limitation of the aircraft. This torque limitation implies that the installed electric motor power cannot be fully used, which is also visible in the maximum thrust chart, where the thrust ratio is limited to approximately 2.5. Changing the design point of the electric motor to the conditions in Sizing No. 3 the maximum available thrust range increases by around 30% compared to Sizing No. 2. Furthermore, the electric motor power gets the dominating constraining parameter in the envelope.

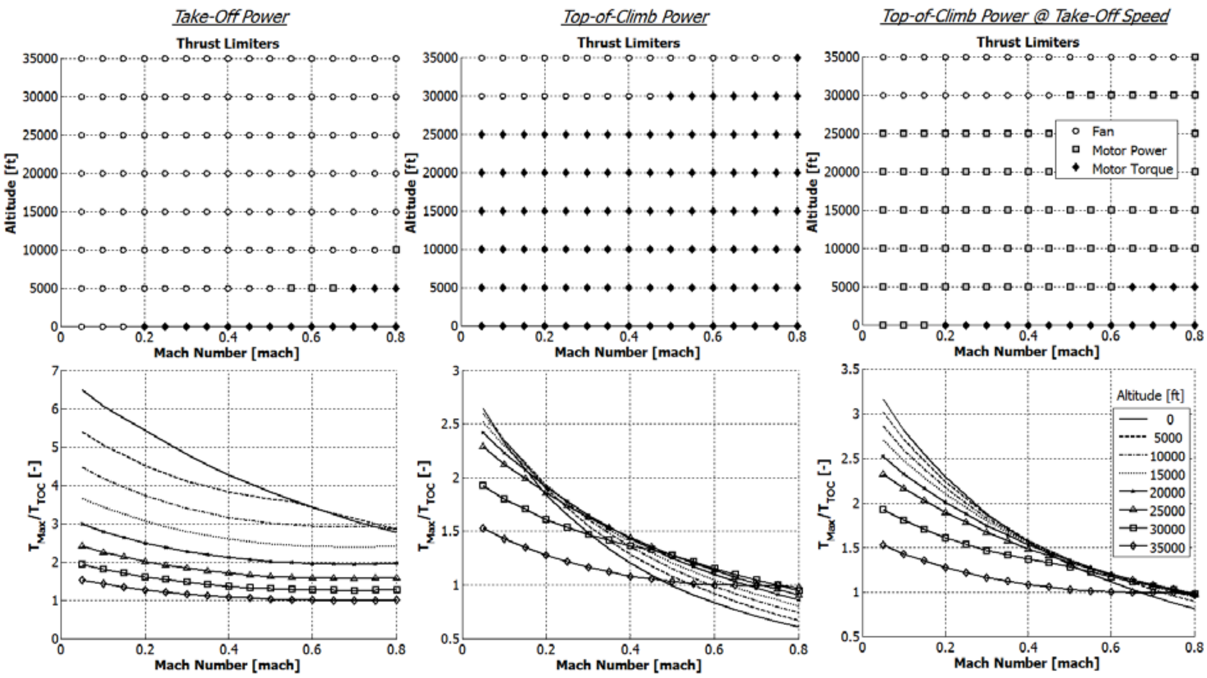

Figure 11: Thrust limiting parameters (top) and maximum available thrust (bottom) based on top-of-climb thrust for different motorization options

5.3 Discrete Parallel Hybrid-Electric Systems Results

In Section 5.2 the focus was limited to the different performance characteristics of the EDF for different motorization options and the maximum thrust levels to be reached for the single design points. Those considerations did not include the ATLeRs. In this section the different motorization options are linked with the gas turbine to fulfil the aircraft TOC as well as the take-off thrust requirements for OEI and one-motor-inoperative (OMI) cases, which is equal to an inoperative EDF. If a thrust requirement is not fulfilled by a motorization option the gas turbine will be sized accordingly to fulfil those requirements by increasing the TOC thrust. Another possibility would be to change the turbine exit temperature T4 during take-off. But this cycle adaption is not considered, as it would decrease the life cycle of the gas turbine. The results of this sizing approach are summarized in Table 4.

Table 4: Thrust characteristics and total mass of the discrete parallel hybrid-electric system

	Sizing No.1	Sizing No.2	Sizing No.3
Gas Turbine Design Thrust [kN]*	16.2	18.8	17.4
Gas Turbine TOC Thrust [kN]*		16.2	
Gas Turbine Take-Off Thrust [kN]**	70.9	82.4	76.4
EDF TOC Thrust [kN]*		6.8	
EDF Take-Off Thrust [kN]**	37.1	12.6	15.6
Total Installed TOC Thrust [kN]*	46	51.2	48.4
Total OEI Thrust [kN]**	145.1	107.6	107.6
Total OMI Thrust [kN]**	178.9	177.4	168.4
TSPC TOC [W/N]	516	503	507
H_P at TOC [-]	0.320	0.322	0.319
Total Mass [kg]	8728	7292	6860
Total Gas Turbine Mass [kg]	4424	5296	4818
Total EDF System Mass [kg]	4304	1996	2042

* Altitude 35,000ft, Ma 0.76, ISA +10°C
** SL, Ma 0.2, ISA +10°C
grey areas mark gas turbine sizing requirement

From an overall hybrid-electric power train perspective it can be seen that Sizing No. 1 leads to a gas turbine sizing at TOC conditions. This is driven by the sufficient available thrust of the EDFs in take-off conditions due to the sufficient installed electric power. The mass of the total EDF system is nearly equal to the gas turbine mass, although the gas turbines are delivering 2.4 times more thrust in TOC. By downsizing the electric system power like performed in Sizing No. 2 and No. 3 the thrust sizing conditions changed from TOC to take-off OEI. For Sizing No. 2 the gas turbine has to be oversized about 16% compared to Sizing No. 1 conditions, which leads also to a 20% higher gas turbine mass. Nevertheless, the EDF system mass reduces by more than 53% resulting in an overall system mass reduction of 16%. The overall system efficiency could be increased by 2.5% in TOC conditions, which is driven by the better gas turbine efficiency. Using the option of the better utilization of the electric system by changing the design point of the electric motor like performed with Sizing No. 3, the gas turbine sizing point can be further reduced by 7.5% compared to Sizing No. 2. This ends up with a gas turbine mass 9.0% lower than for Sizing No. 2 and an EDF system, which is 2.3% heavier. However, the overall system mass could be reduced by 5.9%, while systems efficiency is decreased by 0.8%. Another point, which can be recognized, is that the design H_P in TOC conditions is slightly higher than the reference one. This is mainly caused by using the actual efficiency values for the electric systems architecture instead of an average systems efficiency over the mission. Another aspect is that the H_P varies between the different motorization options. This effect is caused by the gas turbine, as due to the changed sizing points, the gas turbine efficiency varies according to the well-known TSFC bucket curve. But, this H_P change is only marginal, staying below 1%.

6 AIRCRAFT LEVEL FINDINGS

As mentioned in Section 3, the discrete parallel hybrid-electric aircraft concept was in detail investigated by [10]. The overall aircraft level assessment is performed in this study with the help of trade factors to assess Sizing No. 3. For that purpose the reference aircraft has been modelled using an A320 based reference aircraft for the year 2035+ as published by [26] and summarized in Section 3. The modelling of this reference aircraft was performed with the aircraft preliminary design tool PaceLab APD [27]. This tool covers the different aircraft disciplines like aerodynamics and masses with the help of handbook methods, which have been calibrated to meet the A320 mass data. For the 2035+ derivative different aerodynamic, structural as well as system related improvements have been covered as described in [26]. This framework offers also a numerical mission analysis covering the different flight phases like taxi-in and taxi-out, take-off, climb, cruise, descent, reserves like holding pattern and a 100nm diversion phase. Based on those data trade-factors are generated for the change of the system mass including fuel and its implication on MTOW and TOC thrust requirements. Furthermore, a mission thrust profile has been produced, which allows for a decoupled and faster computational analysis of the mission performance to determine the required fuel demand and battery capacity. In contrast to the studies, which were conducted by [10] the hybridization is performed during the entire mission with a constant H_P and not only during climb and cruise. To avoid battery damage due to depth of discharge effects [12] the battery will be sized to a residual SOC of 10% at the end of the overall mission, which corresponds to a 20% end SOC during the block mission excluding the reserves.

The results of this approach are visualized in Fig. 12. It can be recognized that the discrete parallel hybrid-electric aircraft is more than 34% heavier than the reference aircraft at the same mission of 1300nm. This increased MTOW is mainly driven by the battery mass, which accounts for more than 15% of the total aircraft mass. The propulsion system mass has increased by around 68% including the GTF and EDF system as well as the PMAD. Nevertheless, the GTF could be downsized by 2.2% compared to the TOC sizing thrust. Advantageous is the reduced in-flight fuel demand of 3.9% compared to the reference aircraft, which is also equal to a reduction in generated in-flight CO_2 of the same amount, in the first instance. However, the overall energy demand required for the mission has been increased by 14.3%. The overall aircraft mass is 4% heavier than for the hybrid-electric reference aircraft presented in Section 3. This is caused by the changed hybridization strategy, where the electric system is used during the entire mission and not only during take-off, climb and cruise.

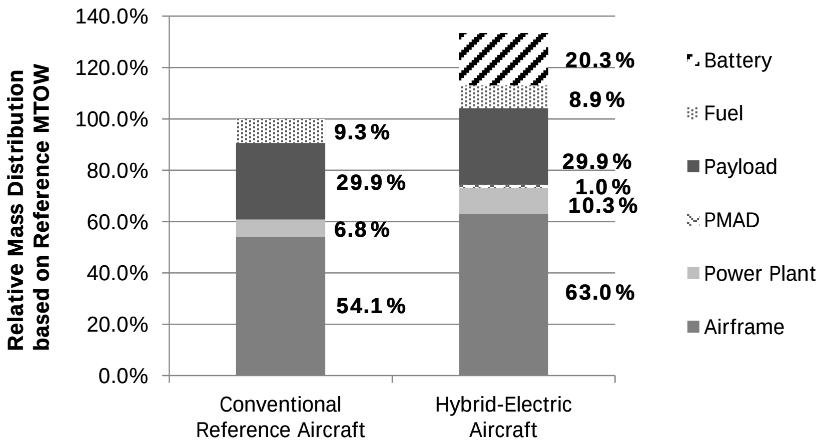

Figure 12: Comparison of the reference aircraft with the discrete parallel hybrid-electric aircraft for 1300nm and a battery specific energy assumption of 1000Wh/kg

Fig. 13 summarizes the results of a sensitivity study using the Sizing No. 3 platform at a H_P of 30% varying only the parameters of the EDF system concerning system efficiency and mass. It can be recognized that a change in the EDF system efficiency has a higher impact on the overall aircraft mass change than influencing only the EDF system mass. This is mainly caused by the effect that an increased efficiency is not only influencing the required battery mass but is also sizing the individual components in the electric system power train, which can be sized for a smaller power requirement. Based on this sensitivity study the following trade-factors for a mass and an efficiency change (given in Eq. 5) can be derived valid for a H_P of 30% at 1300nm design mission

$$\frac{\partial MTOW}{\partial dm}[\%] = 0.0359 \cdot dm[\%]$$
$$\frac{\partial MTOW}{\partial d\eta}[\%] = 0.00413 \cdot d\eta^2 - 0.3485 \cdot d\eta[\%] \tag{5}$$

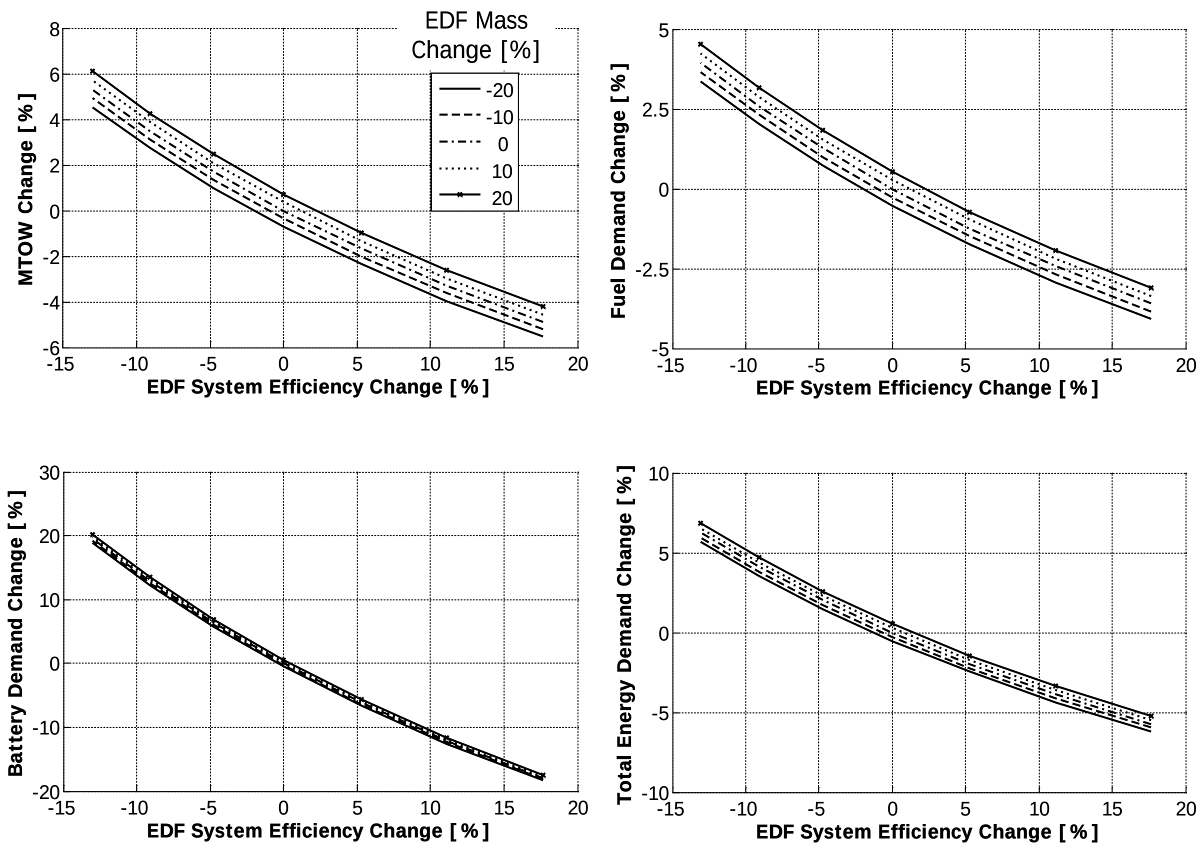

Figure 13: Sensitivity results by changing the overall EDF system efficiency and mass to identify the impact on the MTOW and total energy demand for Sizing No. 3 at H_P 30%

In Section 5.1 a geared EDF was the aim for the performed studies in this paper, because on system level this option leads to a mass optimized system. Based on the derived trade-factors, this design choice can now be evaluated on overall aircraft level and if a direct drive system could be an alternative solution. A direct drive EDF would concern the efficiency to be 0.6% more efficient, but also 22.2% heavier (corresponds to 6.9% total EDF system mass increase including the PMAD) than the corresponding geared EDF with a GR of 2. This would lead to a minor MTOW increase of 0.04%, which indicates also a higher total energy demand. Mark, in this consideration a potential higher nacelle drag, caused by a bigger motor system, is not considered, which can further increase the MTOW and the energy demand. However, if the GR is increased further from 2 to 4, the overall efficiency would decrease by 0.1%, but the EDF mass would also decrease by 12.5% (corresponds to 4.4% total EDF system mass reduction), which would lead to a MTOW decrease of 0.12%. This trend indicates that a geared system could be from an overall aircraft efficiency point of view a potential solution. Nevertheless, concerning the relative low impact on the overall aircraft mass, which is lower than 0.5%, a direct drive system can be an option if beside the emissions also the costs are considered. Due to the simpler design of the system a direct drive system could decrease the maintenance costs of such a system.

7 CONCLUSION AND OUTLOOK

In this paper the impact of different design choices of an electric ducted fan (EDF) with a battery powered electric system architecture have been investigated for a 1300nm fixed wing hybrid-electric aircraft concept. The design target of this study was, how an electric system can be optimally linked to a propulsor and if such a propulsion system can be used to fulfil the environmental targets for the year 2035 specified by the Strategic Research and Innovation Agenda (SRIA). Those studies include the consideration of a direct drive versus a geared power train and the investigation of different motorization options of an EDF, which is used as an assistance propulsion system for a discrete parallel hybrid-electric aircraft concept. In the first study it has been identified that a geared EDF system offers from a mass point of view the best solution concerning an emission reduction potential. The system mass concerning gear box, electric motor, controller and corresponding cooling system can be reduced by 25% up to 30% by varying the gear ratio up to 4 compared to a direct drive system. The system efficiency is only slightly decreasing by 1%. Three different power sizing points are considered to size the electric system architecture for an identical EDF, which are the power demand during take-off, top-of-climb (TOC) and equivalent TOC power at take-off conditions. Using the electric system as assistance system for the gas turbine it could be identified that the sizing option for the electric system at TOC power and take-off conditions offers the highest utilization of the hybrid-electric system. The gas turbine is sized for this scenario to fulfil the aircraft top level requirements during a take-off one-engine inoperative case. With this considered motorization option 21% system mass reduction could be reached compared to sizing the electric motor for the entire EDF take-off power, which cannot be effectively used during the entire mission. Nevertheless, this option represents still a 68% higher power train mass compared to a conventional aircraft propulsion system. On overall aircraft level the hybrid-electric concept results in a 14.3% increased energy demand than the conventional reference aircraft representing the same technology level and mission range. This is mainly driven by the higher maximum take-off weight, which is nearly 34% of the conventional aircraft. Nevertheless, an in-flight fuel burn reduction of 3.9% could be reached in comparison to the conventional kerosene powered aircraft. Those improvements are not sufficient to fulfil the SRIA targets for 2035.

Based on those results it can be concluded that a hybridization strategy, where just the conventional power train is exchanged by a hybrid-electric one, does not offer the best solution with regard to overall vehicular efficiency. For further improvements the flexibility of such new power train concepts has to be used to enable synergies concerning airframe integration and in turn to improve for example the aerodynamic efficiency. Such approaches are currently under investigation to efficiently enable for example technologies like boundary layer ingestion or distributed propulsion, which have the potential to significantly improve the aerodynamic efficiency and decrease therefore the overall mission energy.

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