SYSTEM ANALYSIS OF TURBO ELECTRIC AND HYBRID ELECTRIC PROPULSION SYSTEMS ON A REGIONAL AIRCRAFT

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Abstract

The scarcity of oil and the increasing environmental requirements in the air transport sector pose great challenges for the aviation industry and are a key driver for innovation. Besides various approaches for increasing the efficiency of conventional gas turbine engines, electric propulsion systems have moved into the focus of aviation research. While electric concepts are already in service in general aviation, this study analyses the potentials of electric and turbo hybrid propulsion systems for commercial aviation. The purpose is to compare various architectures of electrical powertrains with a conventional turboprop on a regional aircraft, similar to the ATR 72 on engine and flight mission levels. The considered architectures include turbo electric (power controlled and direct driven), hybrid electric (serial and parallel) and a pure electric concept. Their system weights are determined using today's technology assumptions. With the help of performance models and flight mission calculation the impact on fuel consumption, CO_2 emissions and aircraft performance is evaluated.

Nomenclature

AC	Alternating currency
Alt	Altitude
Comp	Compressor
ed	Energy density
DC	Direct currency
DOH	Degree of hybridization
GT	Gas turbine
HP	High pressure
hy-el se	Hybrid electric serial
hy-el pa	Hybrid electric parallel
IP	Intermediate pressure
LP	Low pressure
Ma	Mach number
MTOW	Maximum takeoff weight
OEI	One engine inoperative
OEW	Operating empty weight
PAX	Persons approximately
pd	Power density
TSFC	Thrust specific fuel consumption
tu-el dir	Turbo-electric direct
tu-el pc	Turbo-electric power controlled
Turb	Turbine
η_{pol}	Polytropic efficiency
Π	Pressure ratio
$\dot{m}_{fuel}$	Fuel flow

1 Introduction

To reach the European Commission's emission goals from Flightpath 2050, which mean 75 % reduction of CO_2 emissions [1] distinct improvements in propulsion systems are necessary. Therefore refining and improvement of conventional turbofan engines is limited and hence not the only option. Besides alternative propulsion systems are focused increasingly by researchers and industry [2] [3]. One possibility is the integration of electric components.

Some small aircrafts in the field of general aviation are already equipped with electric drive systems [4]. In higher power classes electric components do not participate yet in propulsion systems.

This study regards alternative engine configurations applying electric components to a commercial aircraft. The investigations focus

on potential reductions of the environmental impact of a regional aircraft and define influences on gas turbine (GT) cycle and the aircraft.

Based on present (2017) technology assumptions alternative propulsion systems are designed such as turbo-electric, hybrid-electric and pure electric propulsion systems. These are represented in performance models using DLR's in-house simulation tool GTlab [5]. Finally the concepts are compared to a reference case based on flight mission simulation by means of GTlab Flight tool. Besides fuel consumption the emissions and primary energy demands are discussed.

2 Methodology and Application

2.1 Approach

Developing alternative propulsion systems like turbo-electric and hybrid-electric engines implicates an increase of complexity compared to the conventional GT cycle. Since the number of components raises, the total weight and therefore thrust requirements increase. There are plenty of parameters, which influence the aircrafts mission performance. All components must be matched to each other in order to operate the entire system efficiently.

The design of the propulsion systems is based on the individual components technology. Table 1 presents the assumptions for power or energy densities and efficiencies.

Component	pd / ed	Efficiency
Electric motor	5.2 – 9.5 kW/kg	0.95
Battery	200 Wh/kg	0.99
Power electronics	14.3 kW/kg	0.975
DC-DC converter	62 kW/kg	0.989
Cooling system	15 kW/kg	-
Wires	Depending on configuration	0.998

Table 1: Technology assumptions at present state (2017) for design of alternative propulsion systems

The assumptions correspond to state of the art technology levels and are based on a literature review and expert surveys.

According to [6] the power density depends on the surface speed respectively rotational frequency of the electric motor. In each concept it will be adapted to the appropriate level of rotational speed. The electric motors power densities in this study are related to [4] and [7].

The battery is chosen as a high energy cell. Their power density is limited to approximately 0.7 kW/kg [8].

The power density of power electronics is based on manufacturer's data. The presented magnitude is confirmed by [9] [10].

Data for the DC-DC converter is taken from [11].

2.2 Case of Application

This study uses the ATR 72 aircraft as a reference case to determine potentials for electrification. Hence, the propulsion system configurations are designed to meet the requirements for this regional aircraft, which is propelled by two PW127 turboprop engines. Table 2 presents a selection of characteristic aircraft properties for the ATR 72 aircraft model. Typical flight missions for regional aircrafts are around 200 - 300 nm (370 – 556 km), which is much shorter than the maximum range (r), see the Table 2.

Parameter	Value	Unit
PAX	72	-
r	1500	km
v_{max}	510	km/h
MTOW	21,500	kg
OEW	13,000	kg

Table 2: Characteristic properties of ATR 72 aircraft [12]

2.3 Reference case

The different configurations are assessed in comparison to a reference case, which is described in the following. According to the PW127 engine, a DLR127 model is set up within the GTlab performance synthesis tool. This engine operates with three spools, two

compressors, a combustion chamber, three turbines and a secondary air system. Thrust is generated mainly by the propeller, whereas the core engine is used for power generation providing only a small fraction of thrust. The propeller is linked through a gearbox to the power turbine. The layout and station numbering of the model is presented in Figure 1.
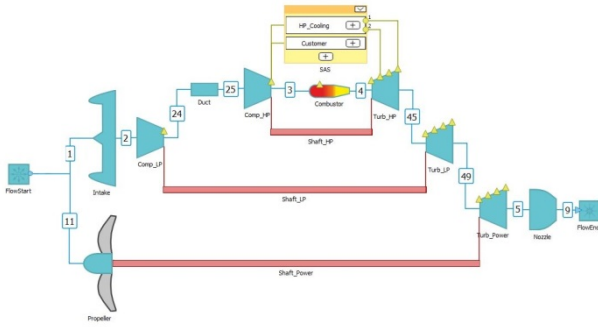
Figure 1: Thermodynamic model of DLR127 engine

For the modelling of the thermodynamic cycle data from literature [13] [14], data sheets [12] [15] and the emissions data base [16] is taken into account. Data could be found for power, rotational frequencies, temperature limits, fuel flow rates and pressure ratios. For engine modelling the cruise operating point is chosen as design point. A selection of performance parameters of the design point are shown in Table 3.

Component	η_{pol}	Π	$\dot{m}_{fuel}$ [kg/s]
Comp IP	0.87	3.4	
Comp HP	0.86	3.4	
Combustor			0.0069
Turb HP	0.84		
Turb IP	0.87		
Turb Power	0.83		

Table 3: Parameters of engine components in design point

For all off-design points compliance of calculated values to temperature limits and plausibility of all other engine data is verified.

3 Electrical Propulsion System Configurations

The alternative propulsion systems are designed in the functional context of the ATR 72. For that purpose properties of the components are balanced and the masses of propulsion systems are calculated as the sum of the components. The requirements for the propulsion systems are derived from the reference engine.

Most alternative propulsion systems use a GT, which principally corresponds to the turbo unit of the reference engine to ensure same technology levels for all configurations. However, to fulfill the different power requirements of the configurations, it is scaled by the mass flow. The principal arrangements of all alternative propulsion systems are shown in Figure 2.
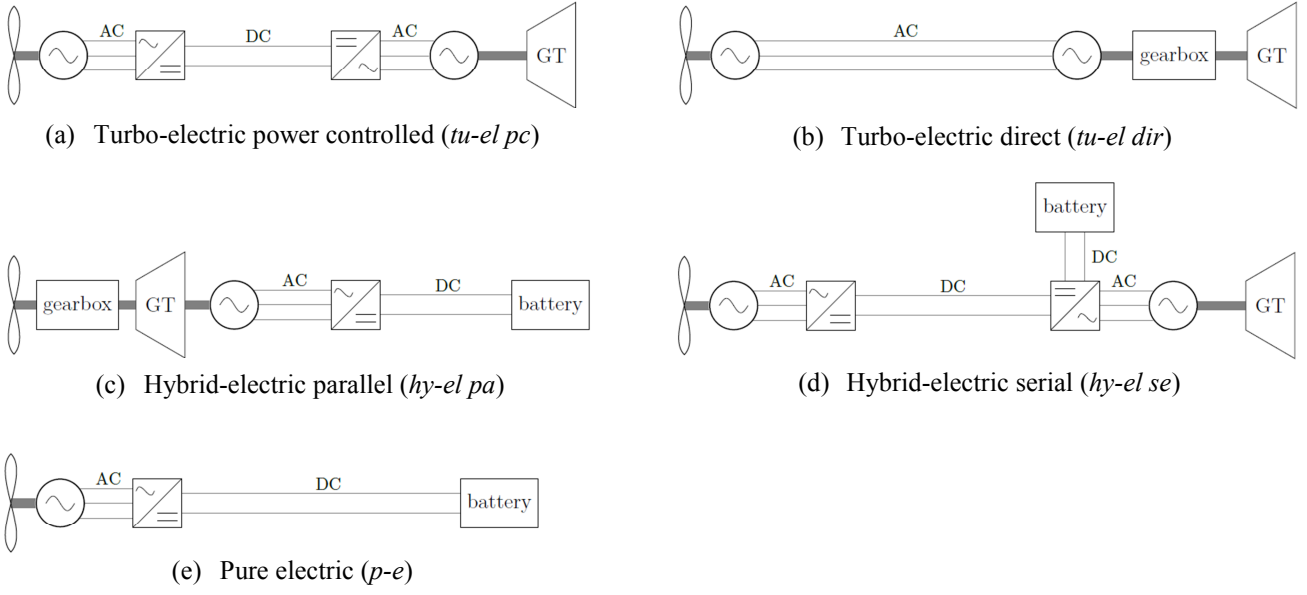
For all concepts the masses are calculated. Table 4 shows the component masses and the effect per propulsion unit. The component masses are based on technology assumptions shown in Table 1. Fuel savings are still roughly approximated here.

3.1 Turbo-electric

In the turbo-electric concept mechanical energy from the GT is converted into electric energy by the generator. Since larger GTs generally work more efficiently than smaller ones, a single GT for both propelling units would be conceivable. The disadvantage in that case is the loss of redundancy for the case of one engine inoperative (OEI). Thus it is not regarded here.

First of all, a turboelectric drive system has disadvantages. Because of multiple converting mechanisms rising fuel consumptions can be expected. However a possible advantage could occur from distributed propulsion units and decoupled rotational speeds of propeller and power turbine. The electrical power distribution may enhance configurational capabilities of the system, which conventional engine design could not enable. In the following two variations of turbo-electric propulsion systems are proposed.

Turbo-electric power controlled (tu-el pc)

Figure 2 (a) shows the principal setup. Between generator and motor two power electronics are arranged. The first one converts the voltage to high DC level reducing conduction losses. The second one creates the voltage and frequency of AC, which is necessary to drive the motor. The

(a) Turbo-electric power controlled (*tu-el pc*)

(b) Turbo-electric direct (*tu-el dir*)

(c) Hybrid-electric parallel (*hy-el pa*)

(d) Hybrid-electric serial (*hy-el se*)

(e) Pure electric (*p-e*)

Figure 2: Principal arrangements of components in alternative propulsion systems

components properties are chosen that way, that a gearbox is not necessary in this concept. The efficiency of electric system yields

$$\eta_{el-sys} = \eta_{mot} * \eta_{PE} * \eta_{cond} * \eta_{PE} * \eta_{gen}$$
$$= 0.8562$$

Due to the number of additional components, the resulting engine is quite heavy. It has more than one ton extra weight compared to the reference engine.
Additional weight in combination with disadvantages in performance will lead to higher fuel consumptions on flight mission level.

Turbo-electric direct (tu-el dir)

Without power electronics the generated electricity is conducted unconverted to the motor (see Figure 2 (b)). That causes a direct link of both rotational speeds. Because of the different rotational speeds of propeller and power turbine there has to be an adaption. This can be reached by a gearbox, a frequency converter, different numbers of poles in motor and generator or any combination of these aspects. Due to Skin- and Proximity-effects a direct conduction with the power turbines frequency is not possible without further changes to the electrical system, e.g. voltage. In order to limit complexity a gearbox is chosen to

be placed in front of the generator. For this purpose the gearbox from the reference engine is assumed to be suitable. That setup results in an electric efficiency of

$$\eta_{el-sys} = \eta_{mot} * \eta_{cond} * \eta_{gen} = 0.8994$$

Although at first glance this configuration seems to be simpler, it finally causes a similar additional weight. Reasons are heavy wires as there is no conversion to higher voltages and the necessity of the gearbox.
This configuration also works less efficient than the reference. On mission based view it is supposed to generate higher fuel consumptions.

3.2 Hybrid-electric

The hybrid electric concepts use energy from two carriers. Additional to the GT a battery delivers energy. In the different phases during flight the energy carriers can be loaded in a way the total system is optimally balanced.
While configuring the battery, besides energy density the power density limitations are also important. The electric support is planned during takeoff and climb. That leads to a smaller GT, because the operating points, which are relevant for sizing of the GT, are electrically supported.

Propulsion unit	tu-el pc	tu-el dir	hy-el pa 10	hy-el pa 40	hy-el se	p-e
Engine	Mass [kg]					
Gas turbine	457.4	451.1	361.5	241.0	411.7	-
Generator	331.8	440.5	-	-	331.8	-
Motor	412.9	412.9	22.6	90.4	412.9	412.9
Periphery						
Gearbox	-	80.0	80.0	80.0	-	-
DC-DC Converter	-	-	3.8	15.0	3.8	37.5
Wires	111.2	275.3	20.5	103.3	111.2	111.2
Power electronics	300.3	-	16.2	65.0	300.3	158.0
Cooling system	19.1	15.2	1.1	4.4	18.3	10.9
Engine total	1,632.7	1,675.0	505.7	599.0	1,589.9	730.6
Difference DLR127	+1,151.0	+1,193.3	+24.0	+117.3	+1,108.2	+248.9
Battery	-	-	292.0	1263.6	292.0	7,742.5
Fuel savings	-	-	-7.5	-30	-7.5	-306,9
Difference total	+1,151.0	+1,193.3	+301.1	+1321.1	+1385.2	+7,377.6

Table 4: Mass effects of alternative propulsion systems per propulsion unit

For reloading batteries at the airport the main influence on loading times is the electric infrastructure. To reach fast loading times it is necessary to have a powerful electricity access. If that cannot be realized, another possibility would be to change the battery.

In the following two different types of hybrid-electric propulsion systems are investigated.

Hybrid-electric parallel (hy-el pa)

In the hybrid-electric parallel setup the electric motor is assembled on the power shaft of the reference engine. The setup is presented in Figure 2 (c). The available space and temperature might be critical points in that arrangement, as the components are located close to each other. The electric motor should not get any extra heat from the GT unit. Therefore it seems reasonable to place the motor on front of compressors or at the gearbox. The electric efficiency for that concept is

$$\eta_{el-sys} = \eta_{mot} * \eta_{PE} * \eta_{cond} * \eta_{bat} = 0.9152$$

Because of the smaller scaled GT the engine is driven with higher temperature in cruise conditions. That behavior is reproduced by the thermodynamic model.

Generating energy during descend is also an option to get enough energy for potential go around scenarios.

The battery is sized for the energy which is required for takeoff and climb.

The scaling of the GT is realized by the selection of the engine design point. It is chosen with lower thrust and power, resulting in a smaller mass flow rate. That corresponds to the smaller scaled engine. Parameters like component efficiencies, pressure ratios or fuel to air ratio are kept unchanged to obtain comparability. However the power coefficient of the propeller has to be adapted, because of the reduced power requirement at the design point. This has to be considered in propeller design of propulsion systems in that functional context.

The amount of additional power from the battery can be chosen freely. For the case of application which is treated here, it is limited by the MTOW of the aircraft.

In the following two variants of parallel hybrid-electric concept are proposed.

_Hy-el pa 10_ supplies 10 % of the takeoff power electrically from the battery. This power stays constant during takeoff and climb. The relatively small share of electric power leads to a moderate battery weight. So the additional weight per propulsion unit is around 300 kg (see Table 4).

_Hy-el pa 40_ has a larger battery and supports 40 % of the takeoff power. The support also

stops after climb segment. The battery becomes so heavy, that the limitations of MTOW are reached with that configuration.

Both variants get close to 0.7 kW/kg power density limit of the battery.

Hybrid-electric serial (hy-el se)

In hybrid-electric serial propulsion system GT and electric propulsion are decoupled. Figure 2 (d) shows that the arrangement is similar to the *tu-el pc* concept. It is notable that in *hy-el se* the generator does not provide all the power. During takeoff and climb power is partly contributed by the battery. Because of MTOW restrictions this cannot be exceeded about 10 % of the takeoff power.

The electric system efficiency during pure generator operating is the same like *tu-el pc* concept $\eta_{el-sys} = 0.8562$. If the battery also provides power, a mixed value will result as total electric efficiency depending on the power ratio of the battery and the gas turbine. Its value will be slightly higher.

Due to plenty of components this concept is complex and heavy resulting in major disadvantages – like in turbo-electric concepts.

3.3 Pure electric (p-e)

The pure electric concept is quite simple concerning its principal setup. It is presented in Figure 2 (e). The battery supplies electric power, which drives the motor. In between power electronics are necessary. Hence the electric system efficiency is analogue to *hy-el pa* concept

$$\eta_{el-sys} = \eta_{mot} * \eta_{PE} * \eta_{cond} * \eta_{bat} = 0.9152.$$

Regarding the component masses the electric motor including all necessary peripheries is already heavier than the reference engine (see Table 4). The battery weight can be seen as the limiting parameter for this configuration, which is not feasible assuming today's technology levels. To enable all electric flight the energy density of the battery has to be roughly 800 Wh/kg. That's four times more than the assumption for today's batteries. Therein, reserves are still not considered yet. At the same time the battery's power density has to be improved by at least the same factor.

3.4 Performance Results

As the performance characteristics of the different propulsion systems have significant impact on the mission fuel consumption, the different behaviors are illustrated in Figure 3. The curves do not include energy taken from batteries in hybrid-electric propulsion systems.
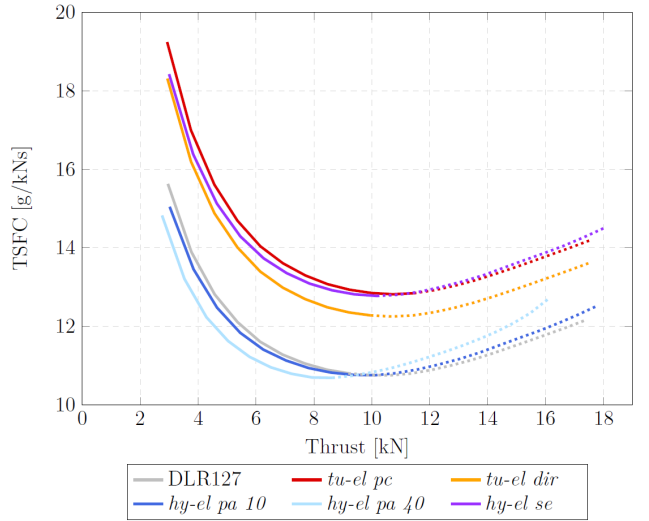
Figure 3: Thrust specific fuel consumption versus thrust of GT in different propulsion systems; dotted lines symbolize that one temperature limit is exceeded (T4 or T49)

The curves confirm that turbo-electric configurations work inefficient because of multiple converting mechanisms. The similarity between *tu-el pc* and *hy-el se* can easily be seen. *Tu-el dir* causes fewer losses, since there are no power electronics. *Hy-el pa* configurations are most similar to the reference engine's behavior. Due to the smaller scaled GT the TSFC-optimum stays on the same level, but moves to a smaller thrust.

The dotted lines indicate the region, where GT temperature exceeds the limits. It can be seen, that the closer the operating point moves to the temperature limit, the more efficient the process works.

The temperature level in the scaled GTs is higher than in the reference. That indicates the necessity of adaption of GT for that application e.g. optimization of the cooling system.

4   Flight Mission Simulation

To determine the fuel consumption on aircraft level the engine data is coupled with an aircraft model via GTlab Flight. Therein, the aircraft is assumed as a point mass model and defined by characteristic masses. Drag polars are based on BADA data. Engine performance data is calculated for the entire flight envelope and different thrust settings and stored within a performance data table. During flight missions altitude, thrust, velocity or Mach number respectively, lift, drag and weight are calculated for each step. The corresponding fuel flow is interpolated from the engine performance data table.

For flight mission simulation in this study a constant trajectory is used for all alternative propulsion systems.

As the aircraft is kept untouched and there are significant differences in the weight of the propulsion systems, it is necessary to reduce the payload to enable constant conditions through all concepts for the compared flight mission. The resulting payload is 4,974 kg instead of 7,000 kg maximum payload of ATR 72. Thus the reference und *hy-el pa 10* configurations transport less payload than they could.

Despite significant differences in the total weights of alternative propulsion units, all concepts are to be integrated into the ATR 72 and transport the same payload. All this has to be respected in design process, which leads to an iterative procedure. The data used in this study is presented in Table 4. It has to be noticed that the presented mass differences apply for each propulsion unit. Therefore the pure electric propulsion system clearly exceeds the MTOW restrictions of ATR 72.
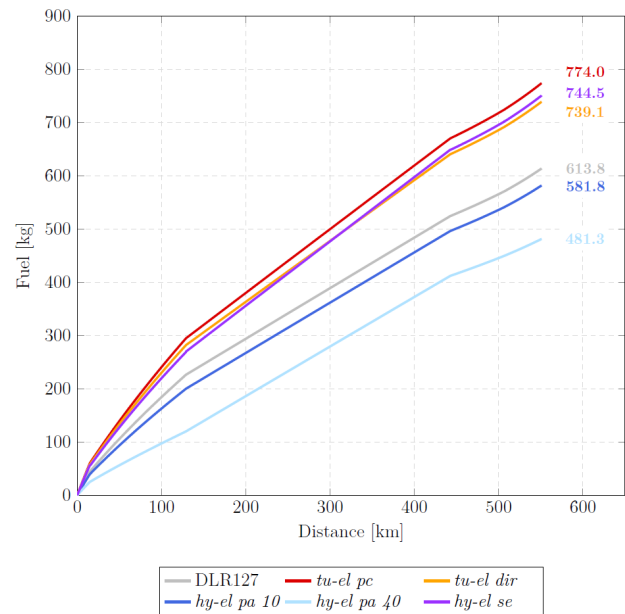
4.1 Fuel Consumption

Figure 4 presents the resulting fuel consumptions on the flight missions with the propulsion systems shown above.

As expected *tu-el pc*, *tu-el dir* and *hy-el se* concepts cause increased fuel consumptions on the flight mission. The difference between power controlled and direct turbo-electric variants can be seen. Additional conversion implicates increased losses and fuel consumption.

Hy-el pa configurations show potential to save fuel compared to the reference engine. During climb the GT is supported by the electric motor. Despite the additional weight of battery this saving can be kept over the whole mission. The almost parallel curves in cruise segment point out that behavior. The saving is 5.21 % for *hy-el pa 10* and 21.6 % for *hy-el pa 40*.


Figure 4: Fuel consumption with alternative propulsion systems and reference engine at 300 nm mission; electric energy from batteries is not presented

To analyze the effect of smaller scaled GTs Figure 5 shows the fuel mass flow rates of the alternative propulsion systems during the compared mission.

In the climb segment the mass flow rates of *hy-el pa* configurations are clearly reduced. In case of the *hy-el pa 40* it is even lower than the level of cruise segment. Here, a design with constant level of fuel mass flow or load in climb and cruise segments seems to be reasonable.

During cruise the *hy-el pa* concepts and the reference configuration are on the same level. Crucial for that is the smaller scaled GT. During cruise it works at an operating point with lower TSFC which is compensated by higher thrust requirements because of the battery weight.
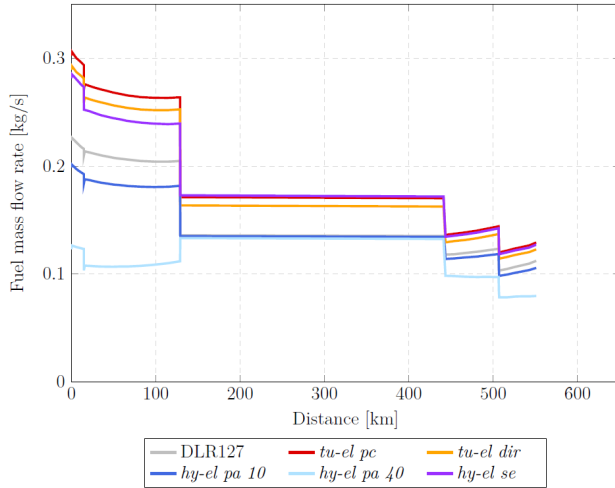
Figure 5: Fuel mass flow rate of different propulsion systems during flight mission

The lower mass flow rates during descend can also be explained by better efficiency at lower load.

4.2 Energy Consumption

The electric energy taken from the batteries was not presented so far. To discuss the total amount of energy, which is necessary to cover the flight mission, the sum of electric energy and fuel bonded energy is shown in Figure 6.
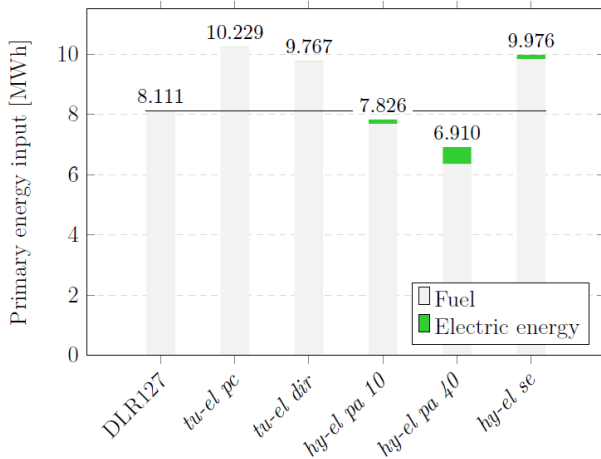
Figure 6: Primary energy inputs for compared mission covered by alternative propulsion systems

A typical fuel heating value for jet fuel of 43.25 MJ/kg [17] is assumed. The shares for fuel and electric energy are each multiplied by a primary energy factor, to consider the efforts to generate and transport energy.

The specific energy input for electricity is 2.13 [18], which is more than the primary energy

factor of kerosene with 1.1 [19]. Thus the share of electricity at the total amount rises.

The optimization of primary energy demands with the help of *hy-el pa* propulsion systems is correspondingly lower. Main influence of the share of electricity is the energy mix in production of electricity. Here it is taken from Germany in 2015.

4.3 Emissions

For a holistic view the behavior of emissions is shown besides fuel consumption and energy demands. Direct CO_2 emissions from combustion of Jet-A1 are 3.156 kg/kg [20]. The equivalent CO_2 emissions are 0.6055 kg/kg [21]. Electricity causes 535 g/kWh emissions according to [22].

Figure 7 presents the calculated emissions splitting direct and indirect ones.
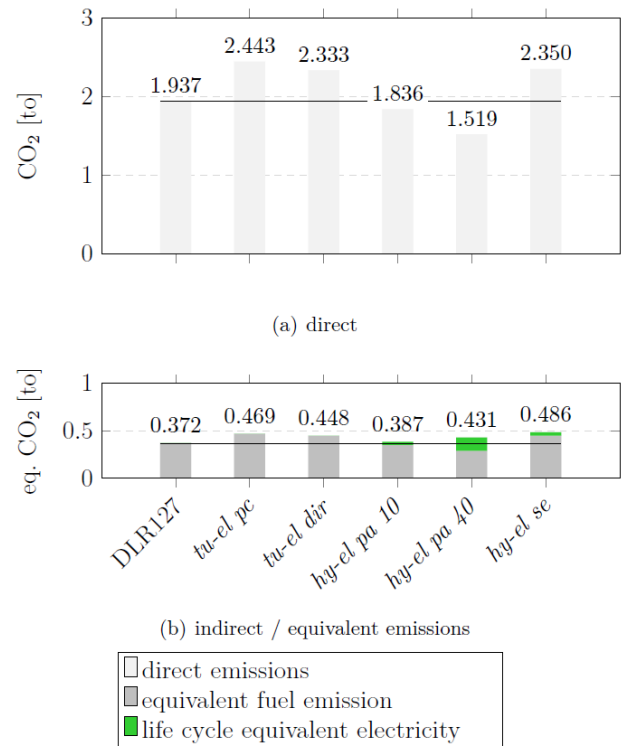
Figure 7: Direct and equivalent emissions for compared mission covered by alternative propulsion systems

The graphic illustrates that the potential for saving emissions is in the area of direct emissions. These are proportional to the fuel consumptions described in section 4.1.

In terms of indirect emissions there is hardly any change for *hy-el pa 10*. For *hy-el pa 40*

there is a slight increase. In general indirect emissions are advantageous as they arise mostly stationary. This offers more possibilities of filtering and treating the emitted gases and particles.

Generally the share of life cycle equivalent emissions caused by electricity can be reduced by an increased amount of regenerative energy in the electricity mix.

5   Discussion

5.1 Weight Analysis

As some parameters in design process are afflicted with uncertainties, their influence on total system weight is examined. This is done by GTlab Stochastics tool. To all parameters a distribution function in form of rectangle, triangle or Gauß is applied. The results are presented in Figure 8.
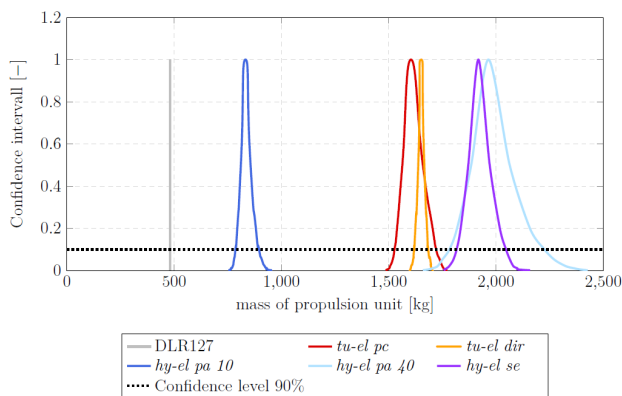
Figure 8: Uncertainties of propulsion unit weights of different propulsion systems

Mass of the reference engine is represented through a line (without uncertainties). The alternative system weights clearly exceed the reference. The peaks of the plots do not exactly correspond to the masses in Table 4. The biggest discrepancy is 5 % for *hy-el pa 40*. The reason is that not all uncertainties are symmetrically distributed around the assumptions.

The dashed line hints the confidence level, which the true total mass will have with 90 % probability. The resulting areas are from different sizes. However, the total masses are

not significantly influenced. The maximum difference to the assumed total weights is around 5%. The main causes for uncertainty are battery and power electronics. Sensitivity analyses prove that their influence increases with raising weight.

5.2 Improvement of the Electric Setup

For hybrid-electric parallel propulsion systems potentials for fuel savings have been identified. Therefore two variants supporting 10 and 40 % of takeoff power during climb have been examined. To analyze in more detail the impact of the degree of hybridization (DOH), which is defined as the share of power provided electrically during takeoff and climb varying DOHs are considered. Figure 9 illustrates mission fuel consumption for different DOHs. It has to be remarked that the result for DOH=0.5 does not fulfill the MTOW requirements of ATR 72 with the payload presented in section 4 because of the battery size. The scaling of GTs implied by variation of DOH is done by reducing the respective design point according to the extra electric power.
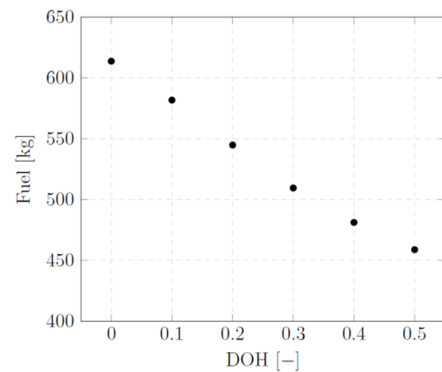
Figure 9: Fuel Consumption of hybrid-electric serial propulsion systems varying DOH on 300 nm mission

The graph in Figure 9 shows that mission fuel consumption decreases almost linearly with rising DOH. At the definition of DOH used here values above 0.5 do not make sense as the climb load of gas turbine would become larger than the cruise load. With further rising charge of electric energy the cruise segment should also be supported electrically. Besides MTOW restrictions temperature is another limiting parameter. Figure 10 presents the temperature behavior depending on DOH.
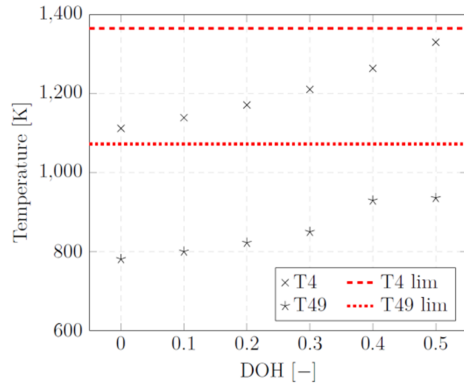
Figure 10: Temperatures T4 and T49 during cruise (Alt=6000m, Ma=0.447) in propulsion systems with varying DOH

It can be seen that rising DOH, which implicates smaller GTs, causes higher temperatures. Temperature limits are provided by [23] for T4. Large DOH move close to the temperature limits in cruise conditions. Warmer environmental conditions and larger loads probably will exceed the limits, which prevents their feasibility. The maximum DOH for that example is 0.4.

5.3 Energy per Payload

The reduced payload represents a restriction for the comparison described above. If ATR 72 is regarded with its maximum payload, it will be considerably harder to reach reductions in fuel consumption. Considering the quotient primary energy per payload none of the alternative propulsion systems installed in ATR 72 reaches the level of reference engine with maximum payload. So it can be concluded that new architectures for aircrafts are necessary respecting the extra weight of electric components.

5.4 Enhanced Technology Level by 2025

To assess the results in terms of the technology assumptions, an enhanced technology level was considered, extrapolating the values to 2025. Therefore, the electric motor power density is increased to about 7.5 – 12 kW/kg with an efficiency of about 0.97. Battery energy density may be enhanced to about 300 Wh/kg, power electronics power density could reach 20 kW/kg

and DC-DC converter are assumed to reach power density of about 80 kW/kg.

These technology levels are applied to the electric configurations and assessed on the flight mission. The fuel consumption results indicate that the *tu-el lg (-7,4%)*, *tu-el dir (-4,6%)* and *hy-el se (-7,4%)* configurations are improved significantly. However, the *hy-el pa 10 (-0,4%) and hy-el pa 40 (-3%)* systems show lower fuel consumption benefits. Hereby, the lighter battery system may increase the payload capability and hence, enhance the energy per payload ratio.

6   Conclusion and Outlook

Currently known electric aircrafts are – mostly because of energy storage – clearly limited to short distances and low power requirements. The analysis of propulsion system weights shows that electrification for commercial aircrafts just works with turbo-electric or hybrid-electric propulsion systems with current state of the art technology. These have been simulated on flight mission basis with the example of an ATR 72 aircraft. Turbo-electric aircrafts both – power controlled and directly driven – generate massive extra weights and loss of total efficiency. The same goes for the hybrid-electric serial propulsion system which is mostly similar. Further development of these configurations becomes viable, when superconducting technology is getting closer. Potential gains due to distributed propulsion are to be exploited then.

Pure electric propulsion systems need a massive improvement in energy – and power density of batteries. The regional flight mission regarded in this study (with reduced payload) requires at least a further development by factor 4 for both. Parallel hybrid-electric propulsion systems have a potential to reduce fuel consumption on flight missions. The savings are realized by a changed gas turbine design in combination with an electric motor driven by a battery. Peak loads are covered by the electric system and may lead to increased gas turbine temperatures. With rising size of the electric system up to 21 % savings in fuel consumption are feasible. Direct emissions behave proportional. Indirect

emissions stay on a similar level like the reference case. Their share could be reduced by an increased amount of regenerative energy at the electricity mix.

For regional aircrafts potential for fuel - and emissions savings have been proved. A moderate extrapolation of the technology level may enhance the fuel flow benefits and underline the general trends. However, due to the weight penalties electric propulsion systems require a modification of the aircraft architecture. The restriction to short ranges may be overcome trough new batteries in the future.

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