SAFETY-DRIVEN BASELINING OF HYBRID ELECTRIC AIRCRAFT ELECTRICAL POWER SYSTEM ARCHITECTURES

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Abstract

Electrification of aircraft power and propulsion systems is critical for reduction of aircraft emissions (greenhouse gases and acoustic noise). The disruptive nature of electrical propulsion systems for aircraft, and the associated lack of legacy electrical power system (EPS) solutions presents as an opportunity for new solutions to optimize the overall performance of these new aircraft. However, the lack of legacy architecture solutions, combined with the increased power levels of hybrid electric aircraft, is a major challenge to the design of EPS to meet performance requirements (reliability, weight, volume, efficiency).

Existing approaches to EPS design for aircraft with electrical propulsion (all or hybrid) assume a starting point with a comprehensive set of baseline requirements, sufficient to commence informed EPS design. This paper directly addresses the challenge of how to determine these baseline requirements for a new concept aircraft with minimal initial design criteria, and ensure that architectures developed will meet safety requirements. This is achieved by translation of expected certification criteria to failure modes, during the occurrence of which flight must be maintained. From these, baseline requirements for subsequent EPS designs, including system trades for optimized EPS architecture solutions and interfaces with non-electrical power systems, are captured. Through a case study for a concept, low emission distributed, hybrid electric propulsion, long aspect wing ratio aircraft, capture of baseline criteria and subsequent EPS design (including system design trades) and interfaces to associated non-EPS systems design is demonstrated.

Keywords: Distributed propulsion aircraft, aircraft electrical power systems

1. Introduction

The pathway to net zero aviation is via a combination of new technologies, including fuels (hydrogen and sustainable aviation fuel (SAF)), new wing designs (high aspect and blended), new engines (ultra-high bypass and open rotor) and increased electrification of on-board power and propulsion systems [1]. These interdependent technological advances combine to provide significant efficiency benefits (20 - 40% [2]), reduction of greenhouse gas emissions and acoustic noise. The flexibility of physical location of electrical propulsors increases aerodynamic efficiency by ~ 12% [3]. However, the weight and volume of the electrical power system (EPS) must not mitigate this efficiency benefit, and must maintain sufficient resilience for this safety critical application.

The design of the EPS for these new aircraft is challenged by the low technology readiness level (TRL) of the EPS and associated equipment; the lack of legacy, commercially operating systems and the need to develop these technologies within a short timeframe to meet targets such as those set by the European Commission's Flightpath 2050 [4]. Additional challenge is presented by the interdependencies of the EPS design with non-electrical power systems, in particular the wider propulsion power train, and how to incorporate these into the EPS design. Further, the safety critical nature of aviation necessitates that the aircraft must be able to maintain flight in the event of equipment or sub-system failures. Hence the propulsion power train must be designed to accommodate failure conditions, including the EPS for failure conditions [5]. As a consequence, decisions made for EPS architecture design are driven by approaches to prevent catastrophic failure of the propulsion system.

This paper addresses the challenge of design of an EPS for a distributed hybrid electric propulsion (DHEP) concept aircraft, where extremely limited information is available to support the design of a baseline EPS. The aim of this concept aircraft is to minimize emissions (greenhouse gas and acoustic noise) at low altitudes (below 900 m) near airports [6]. The minimization of emissions is proposed to be through the electrification of propulsion as far as possible, combined with use of sustainable aviation fuels. The aircraft will have two turbo-props, each driven by a gas turbine engine, and a number of electrically driven propellers (e-propellers) along each wing.

Due to the interdependencies of the EPS with non-electrical systems, it is critical that a systems level approach, rather than a technology level approach (such as that presented in [7]) is taken. This is to enable efficient, early stage identification of solutions which meet required performance targets (reliability, weight, volume, efficiency) and to carry out key systems design trades, both for the EPS and non-EPS, systems. Methods for performance analysis of EPS for concept DHEP aircraft are presented in the literature, e.g. [8-10]. However, these have a starting point of a baseline EPS.

A hierarchical methodology (Fig.1) for de-risked EPS design, for cases where no baseline EPS exists as a starting point for an all or hybrid electric aircraft, is presented in [11,12]. This heuristic approach was driven by a combination of the systems design "V" [13], and the SAE AIR 6326 modelling paradigm [14]. To derive an initial EPS architecture, the methodology takes in a set of baseline requirements at "Level 0", which are: a mission profile with electrical power demand at each stage of flight; number of electrically driven propellers; type and number of electrical power sources; number of electrically driven propellers and electrical power sources which can fail; and selection of DHEP topology. By inclusion of the electrical power levels, decisions on level of hybridization between electrical and non-electrical propulsion are inherently made. The methodology subsequently systematically works through a set of system design levers (SDLs) at Level 1, to derive a candidate EPS. Candidate architectures are assessed against performance requirements (weight, volume, reliability, cost), prior to undertaking transient studies (at Levels 2, 3 and 7) to verify system functionality and further refine the architecture design. Datasets are held in Levels 4, 5 and 6.

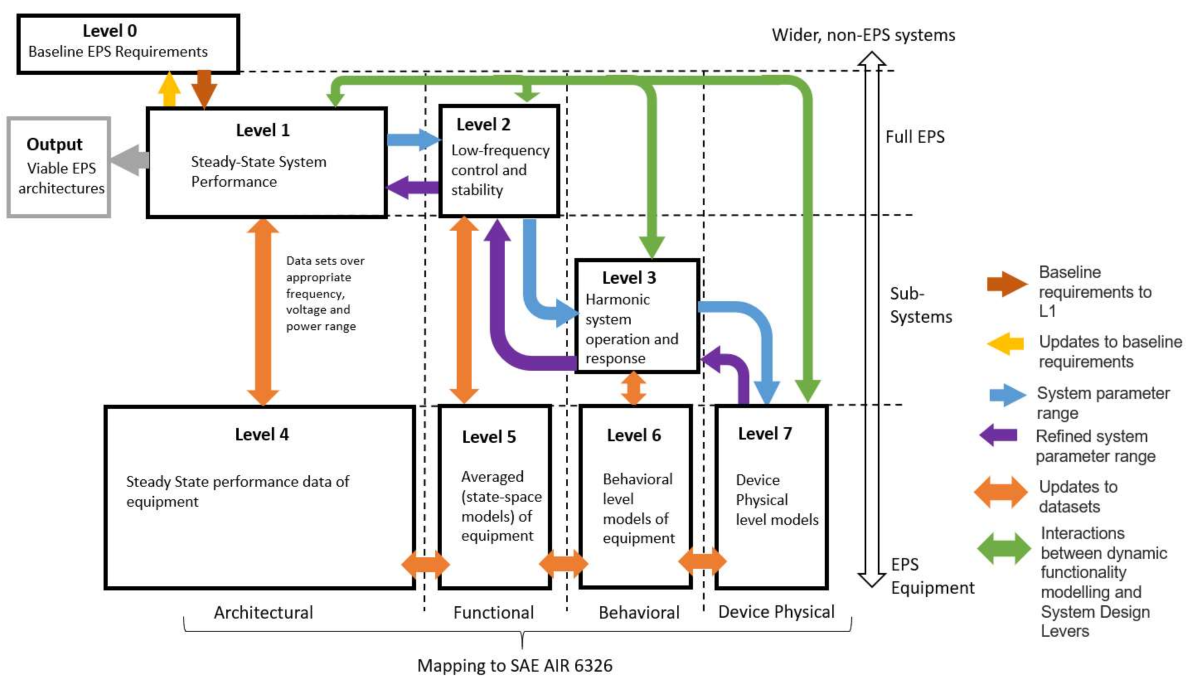

Figure 1 – EPS Modelling Design Framework for design of EPS for disruptive applications [11,12].

A significant limitation of the EPS modelling framework in [7,8], is that the starting point assumes the set of baseline requirements are decided. This limitation is particularly pertinent when undertaking the design of an EPS for a new concept aircraft where limited information is available. This paper addresses this gap by presenting a methodology to capture these baseline requirements at Level 0, and updating the process at Level 1 to identify candidate EPS architectures which can mitigate failure modes, which would otherwise lead to catastrophic failure. The safety critical nature of air travel, necessitates that aircraft systems design must incorporate safety considerations from an early stage in the design process. Due to the close coupling of the EPS with the wider propulsion system, the methodology must also incorporate design interdependencies with the non-electrical propulsion system, and where appropriate, wider aircraft design. The rest of this paper is presented as follows: Section II presents the proposed approach to capture baseline requirements, and revised approach to EPS design; Section III applies this methodology to a concept low aspect wing ratio DHEP aircraft EPS with design trades on approaches to mitigate failure condition, degree of hybridization and electrical power split between electrical power sources. Section IV presents discussion from the results and finally Section V presents next steps and future work.

2. Proposed Methodology Incorporating Capture of Baseline Requirements

2.1 Overview of Design Process at Level 0 and Level 1

In the EPS Modelling Framework (Fig. 1), the baseline requirements required as inputs to Level 1 to enable EPS design are the high level DHEP topology, number of sources and loads, choice of electrical power sources, mission profile (power requirements and duration) and information regarding numbers of loads and electrical power sources which can be lost, but flight still maintained. For an electrical propulsion system, electrical loads include motors for electrically driven propellers (e-propellers) (series hybrid), or provide electrical support to the gas turbine shaft (parallel hybrid) [5,15].

The hybridisation factor (box “1” in Fig.2), *H (%)*, is a further input from Level 0. This is defined as

$$H = \frac{P_e}{P_e + P_{tp}} * 100 \qquad (1),$$

where P_e *(W)* is the propulsive power from the e-propellers, and P_{tp} *(W)* is power from the turbo-prop. The level of H is a critical interaction with the non-electrical systems in the propulsive drive train, directly impacting on the sizing of the gas turbine, turbo-prop and e-propellers. As such, limits for upper and lower levels of *H* may be set by the non-EPS system design.

The revised design process for Levels 0 and 1 (Fig. 2), enables derivation of these baseline requirements, driven from a safety case perspective. The number of electrical power sources and loads is strongly driven by interdependencies with non-electrical power systems. For example, number of gas turbines and aerodynamic limits for unsymmetrical propulsive power from propellers (turbo-prop or e-propellers).

The number of sources and loads are also interdependent with the derivation of baseline safety cases. It is proposed that the baseline safety cases must be determined from failure modes, which are derived from Certification Specifications and Acceptable Means of Compliance (box “2” in Fig. 2). Where these do not exist for the concept aircraft of concern, expected failure modes can be inferred from existing regulatory documentation. Hence the translation to failure modes (box “3” in Fig. 2) combines the hybridization factor, choice of power sources, Certification Specifications and Acceptable Means of Compliance.

At Level 1, baseline information from Level 0 of high level DHEP topology (e.g. series, series/parallel, turbo-electric, parallel) and mission profile is used to capture power requirements at each stage of flight. The failure modes derived in Level 0 are used to determine assumptions for the EPS design to ensure flight is maintained, resulting in consequences for the EPS design (box “4” in Fig. 2). Example outcomes from this stage include decisions on interconnection, or number of sources required per channel. The assumptions and consequences lead to identification of trade studies to refine and optimize the EPS architecture (box "5" in Fig. 2).

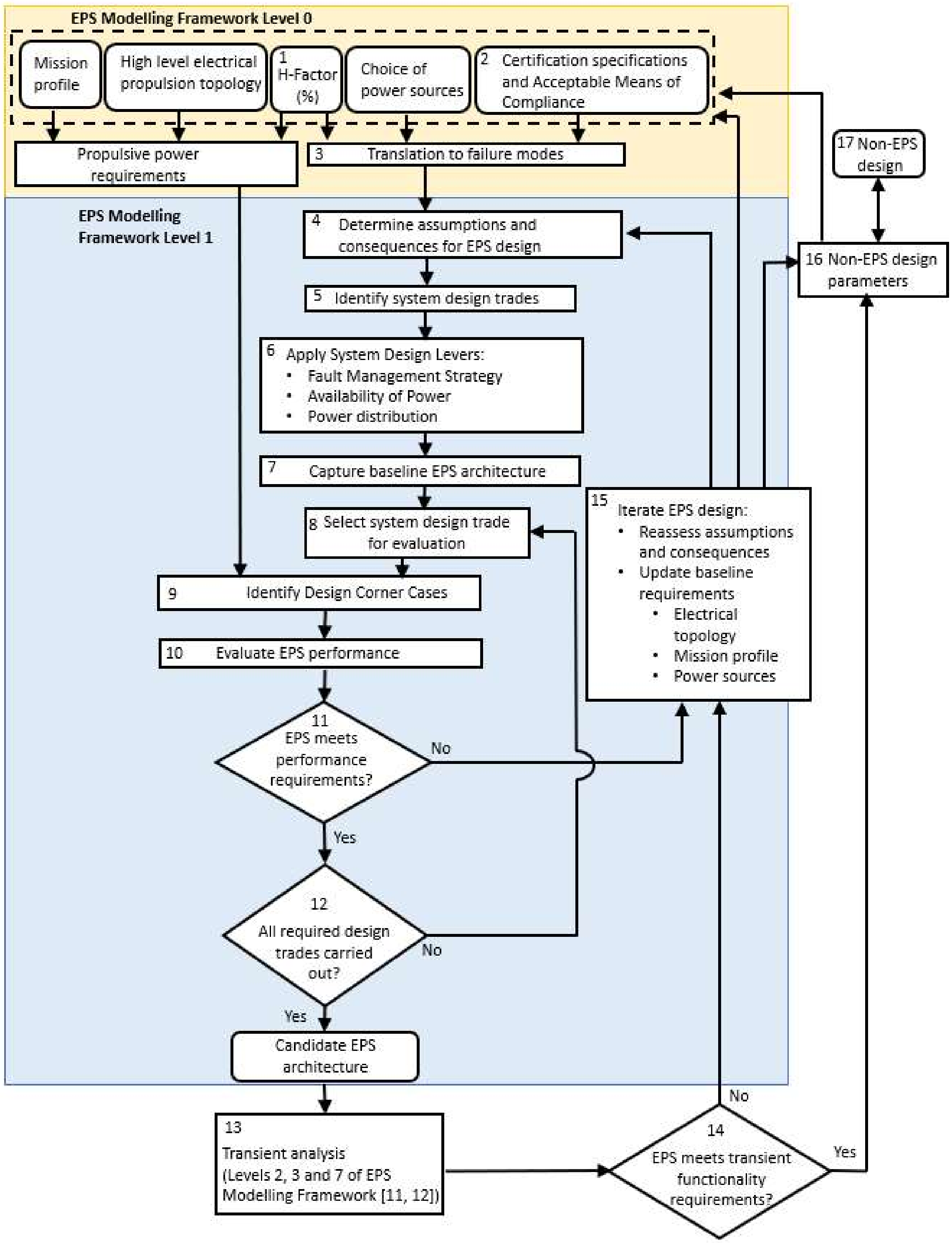

Figure 2 – Methodology to capture baseline requirements and subsequently derive a viable EPS architecture for a DHEP concept aircraft.

At this stage, the SDLs, described in [11] to generate an architecture for an EPS, are incorporated into this revised process (boxes “6” and “7” in Fig.2) As was discussed in [12] for a preliminary design, the SDLs of Fault Management Strategy (FMS), Availability of Power and Power Distribution are considered. The SDLs of Power Quality and Thermal Management are not considered at this stage, but may be considered at a later stage to fully optimize a solution. Baseline requirements are worked through the SDLs to derive an initial candidate EPS architecture which will mitigate failure conditions. This is not a final EPS design: the set of trade studies identified in box “5” in Fig. 2 must be worked through to refine the architecture.

To select an initial system design trade to evaluate EPS performance against (box “8” of Fig. 2), a failure rate analysis of the proposed components of the EPS is carried out to assess where failure rate bottlenecks are likely to occur within the proposed. This is used to prioritise system design trades. An initial system design trade is selected to evaluate against the performance of the EPS. Design corner cases of the EPS architecture are defined (box “9” in Fig. 2), for example maximum power demand under nominal and off-nominal (failure) conditions. The propulsive power requirements are incorporated from Level 0 at this stage. During this stage, alongside incorporation of the H-factor, the power split between different electrical power sources will be included. Performance evaluation (box “10” in Fig.2) includes assessment of weight, efficiency, losses, volume and cost. More detail of the evaluation process is given in Section 2.2.

If the outcome of the evaluation process is that the EPS does not meet performance requirements (boxes “11” and “15” in Fig.2) then iteration takes place to change the baseline architecture. Depending on why performance has not been met, iteration may be of assumptions made in box “4” of Level 1; design of non-electrical power systems to adapt baseline requirements, including mission profile, choice of power sources or DHEP topology; or the design of non-electrical systems in the drive train, or wider aircraft.

If a candidate architecture has acceptable performance, further trade studies are carried out to refine the EPS design (boxes “12” and “8” in Fig.2). Once all trade studies have been concluded, and the EPS meets performance requirements, then the transient functionality of the proposed solution is assessed via time-variant transient analysis at Levels 2, 3 and 7 of the Modelling Framework (boxes “13” and “14” in Fig.2). If the EPS requires adaptation to function, then baseline parameters (for example choice of power sources), or SDLs are updated as appropriate. If the failure of the transient functionality is due to interaction with the non-electrical propulsion system, then this must be fed back and non-electrical propulsion system iterated. Once a viable solution is found, this is fed back to the non-EPS design (boxes “16” and “17” in Fig. 2).

2.2 Evaluation of EPS Performance at Level 1 of EPS Modelling Framework

The performance analysis tool used within the EPS Modelling Framework is summarised in Fig. 3. The tool has been implemented in Python. The tool takes in the EPS architecture, mission profile, range of hybridization factor, *H*, and electrical power split between different electrical sources. Within the tool, this information is used to populate a network configuration file, and the EPS is represented as a network of nodes. The component database is used to populate the network with relevant data, such as power density or efficiency. The tool splits the aircraft mission into specific fight phases under nominal conditions, and performs power flow calculations for each phase of flight. Flight phases are differentiated by the power demand of the EPS. The tool sizes the electrical equipment and components required for each flight phase, based on the power flow and the predefined power density and energy density for each component type.

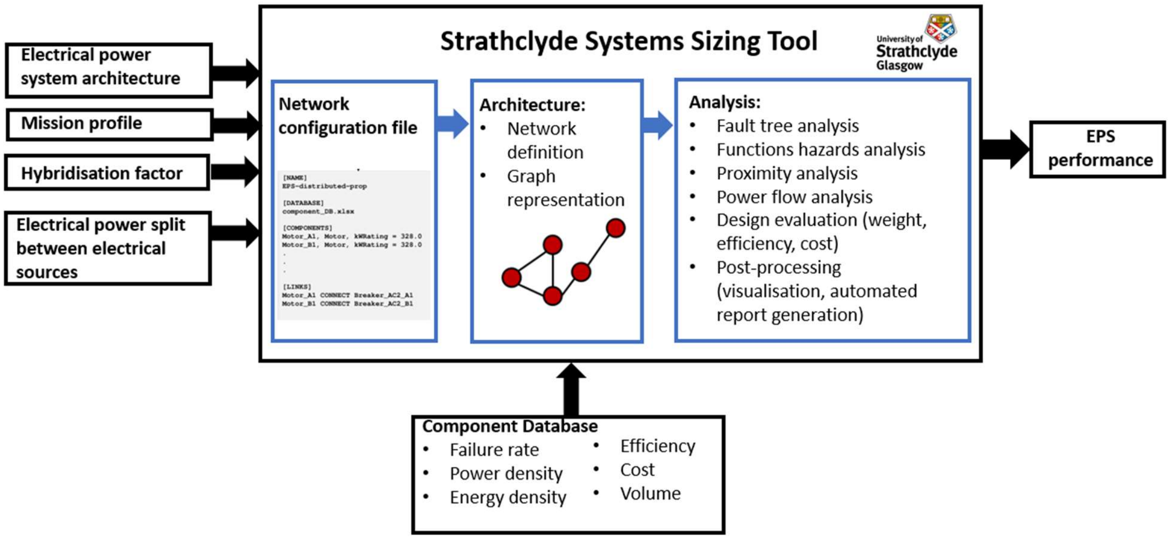

Figure 3 – Overview of the Strathclyde Systems Sizing Tool for assessing EPS performance.

To obtain the weight, volume and cost of a candidate EPS architecture, the tool assesses the weight of components for each of the design corner cases by carrying out power flow calculations to size EPS equipment. The heaviest estimated weight, largest volume or highest cost for each component is used to estimate the total weight, volume or cost of the EPS. Electrical losses and efficiency are calculated for each mode of operation.

The weight of generators, motors, power electronic converters and solid state circuit breakers is based on power density from the published literature, as is energy density for energy storage. The busbar is sized by calculating the required cross-sectional area to provide sufficient current carrying capacity, with the assumption that the busbar is copper with a current carrying capacity of 1.2 A/mm^2. The current level is calculated using the power flow and selected voltage level. Hence the weight of the busbar is then estimated from the calculated cross-sectional area, an assumption that the length is 0.5 m and the density of copper is 8.96 g.cm^3. The sizing tool estimates the location of components, and from this the tool extracts the length of cables within the EPS. The current rating of the cables is calculated from power flow and pre-defined voltage levels. Combining current density 170 A/(kg/m) [16], current level and length of cables, the weight of the cables is estimated.

3. Case Study Demonstrating Safety-Driven Baseline Architecture Capture

3.1 Capture of Baseline Requirements at Level 0 of EPS Modelling Framework

The concept aircraft that is the focus of this case study is a DHEP aircraft, with circa 60 pax, of similar size to an ATR72 aircraft [17]. Propulsion for the aircraft is provided by a combination of two turbo-props, one per wing, driven by a gas turbine, and 4, equally rated and physically sized e-propellers per wing. From a review of the literature, the baseline EPS topology is chosen to be a serial/parallel topology [5], with generators (unspecified number at this stage) driven by the gas turbine. The use of batteries alongside the generators to provide power below 900 m, is to be investigated as an option for reducing emissions at low altitudes. The power split between generator and battery is not specified, nor is the hybridization split between e-propellers and turbo-prop propulsion. At a later stage (outwith the scope of this paper) the possibility to support turbo-prop operation by running the generators as motors, powered by the battery will be investigated. The mission profile is presented in Fig.4.

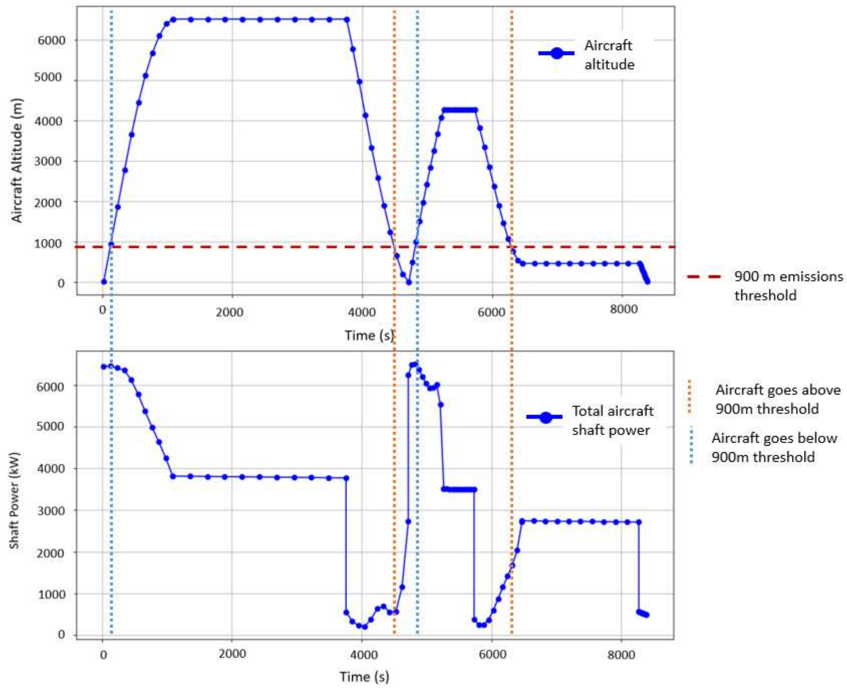

Figure 4 – Baseline mission profile for concept aircraft, with indication of 900 m threshold and associated power levels. Includes missed approach and subsequent go-around after 4,800 s. Aircraft altitude against time (above). Total aircraft shaft power against time (below).

Certification of DEHP aircraft is challenged by the disruptive nature of the technologies, and the non-applicability of existing certification regulations to these new technologies and systems [19]. However, by taking a high level view of the Certification Specifications and Acceptable Means of Compliance [20] for large aircraft with conventional gas turbine driven propulsion, and the limited Special Conditions available for electrically driven propulsion e.g. [20] and [21], a set of failure modes for the proposed concept aircraft can be captured.

Table 1 presents the Certification Specifications and Acceptable Means of Compliance identified as relevant to the power train design of this concept aircraft. Following the methodology presented in Section 2, failure modes have been identified from the Certification Specifications and Acceptable Means of Compliance, and described in Table 1. A key failure mode identified is loss of electrical power sources (generators, batteries) and propulsive power sources (turbo-prop, e-propellers). If the failure rate of a gas turbine is assumed to be 2.67 x10^{-6} failures per flight hour [18], then it is acceptable to assume that the design must accommodate the failure of one engine, rather than both engines. An assumption is made that the wider aircraft design, such as the aerodynamics, is such that flight can be maintained under this one engine inoperable (OEI) failure mode. The number and location of e-propellers which can be inoperable, and flight maintained, is a decision which must come from the wider aircraft design. To provide an example for this case study, it is assumed that 6 out of 8 e-propellers must be operable, and that 2 e-propellers on one wing can fail and flight maintained.

Regulation Documentation	Summary	Failure Modes
CS 25.1309 [20] SC E-19 [21]	Single failure does not result in catastrophic failure	Loss of • gas turbine • all e-propellers (due to common cause failure) • electrical power sources other flight critical systems.
AMC 25.981(a) [20]	Must have adequate electrical protection to prevent electrical fault resulting in ignition of fuel. Method must be failsafe.	
MOC-2 SC-VTOL [22]	Must be able to disconnect and isolate a battery quickly. If disconnected manually, must be able to reconnect in flight.	Loss of battery
AMC 25.1351(d) [20]	Must have back-up electrical power for electrically driven, flight critical systems.	Loss of electrical power sources and non-electrical propulsion.
AMC 25.671 [20]	All engine failure: sufficient emergency power to power all control systems.	
CS 25.1351 [20]	Maintain voltage and frequency.	Loss of electrical power sources and loads.
	Transients do not result in a hazardous situation.	Transients due to protection tripping.
	Power sources operate when interconnected and separate	
SC E-19 [21]	Must be able to restart engine mid-flight	Loss of electrical power sources and non-electrical propulsion.

Table 1 - Translation of regulatory requirements to failure modes

3.2 Design Considerations at Level 1 of the EPS Modelling Framework

3.2.1 Evaluate Impact on Design Considerations for the EPS

From the failure modes identified and summarized in Table 1, the resulting assumptions and impacts on the EPS design are captured in Table 2. At this stage the full functionality of the EPS is considered, including approach to electric engine start and supply of power to the e-propellers.

The first consequence of the assumptions presented in Table 2 are that the generators are connected to the low pressure (LP) shaft of the gas turbine, with starter motors connected to the high pressure (HP) shaft. For redundancy at this stage, it is assumed that the battery supply for the electric engine start motor and for direct support of the electrical propulsion system, are separate. The failure rate of a generator is 1.3×10^{-4} failures per flight hour [23]. The failure rate of the gas turbine is lower than the failure rate of a generator by an order of 2 magnitude. Hence 2 generators are connected to each LP shaft of the gas turbine. A second consequence of these assumptions is that capability for interconnection between channels is required to enable electric engine re-start mid-flight by utilizing electrical power from generators driven by the operating engine, and for engine start on the ground if a ground power unit (GPU) is used to start one engine, which subsequently provides power to start the second engine. Batteries could be used to provide this redundancy. However, interconnection provides redundancy with a lower weight penalty than is associated with additional battery capacity.

Failure mode	Assumptions	Consequences	Trade studies

Loss of a gas turbine engine	• Generators connected to LP shaft.
	• Implement electric engine start:
	  • Starter motor connected to HP shaft
	  • Engine starter motor cannot be driven from generator on the same engine.
	  • Provide power to starter motor from generator on the other engine.
	  • One starter motor per engine
	  • Additional supply of power to starter motor for case of double engine restart required.
	  • Electric engine start on ground using GPU due to no APU.
		• Separate electrical machines for generators and motors.
		• Require interconnection in EPS between engines.
		• Reduces sizing of starter-battery.
		• Battery must be sized to enable three attempts at electric start.
		• Require electrical interconnection between generators for GPU, and electric starter motor driven by electrical power from other engine.
			• Options to provide redundancy:
			  • Oversize generator, battery or turbo-prop.
			• Approaches to interconnection.
			• Electrical interface to GPU:
			  • 115VAC or DC.

Loss of a generator	• 2 generators per LP shaft if the failure rate of 1 generator is higher than the engine failure rate.	• Two generators per engine.	• Location of protection devices. • One generator if dual wound machine.
Loss of all generators, unable to restart engines.	• Use propeller motors to provide RAT functionality.	• Electrical protection devices must be resettable. • Downstream devices must allow some degree of bi-directional power flow.	• Role of battery if all generators are lost.
Loss of power source.	• Each generator has own bus • Battery connected to same bus as generator	• Interconnection between channels is required. • One battery and generator per channel. • Able to isolate generator and battery upstream of bus.	• Number of channels • Approach to interconnection. • Physical location of the bus • Battery failure: • Oversize batteries, generator or turbo-prop for redundancy.
Failure upstream of generators	• Generators are permanent magnet synchronous machines • Able to detect and isolate faults.	• Need to manage current under fault conditions. • Appropriate positioning of protection equipment and sizing of e-propeller motors.	• PMSM versus use of dual wound synchronous generator. • Location of disconnection points.

Table 2 – Derivation of assumptions, consequences and trade studies for the EPS design from identified failure modes from Table 1.

3.2.2 Application of the System Design Levers

The baseline requirements indicate a system with 4 generators, 4 batteries and 8 propulsor motors to drive the 8 e-propellers. From [11,12] it has been established for the capture of a preliminary EPS design, the critical SDLs, alongside baseline requirements and identification of design corner cases, are FMS, Power Availability and Power Distribution. Power Quality and Thermal Management are used to fine tune an EPS at a later stage. The decisions made are summarized in Table 3, with a more detailed explanation provided below.

The FMS requires adequate redundancy in the event of the failure of up to 2 generators and up to 2 motors. This necessitates that for the case of the loss of an engine or batteries, power sources are sized to provide sufficient power for flight. This can either be achieved by oversizing electrical power sources (generator, battery), or by sizing the gas turbine and turbo-prop to meet the additional power demand, in the case of an OEI condition. For loss of e-propellers, the AC-DC converters and propulsor motor (forming an electrical propulsion unit (EPU)) used to drive each e-propeller, and each e-propeller, must be oversized as appropriate.

System Design Lever (SDL)	Design Requirement	Consequences
Fault Management Strategy	• Adequate redundancy in case of failure of up to 2 generators and 2 EPUs.	• Power sources sized to maintain flight during failures: • Oversize EPUs and e-propellers or turbo-prop. • Oversize electrical power sources or turbo-prop.
	• Ability to isolate electrical faults.	• 4 channel system • Each generator and battery has its own bus • Zonal protection scheme. • Resettable protection devices.
Power Availability.	• Maintain power to all healthy EPUs in the event of a fault.	• Capability for interconnection between channels. • Interconnection supports electric engine start from one engine to another.
Power Distribution	• Electrical decoupling between generators and motors. • Ease of interconnection	• DC distribution for electrical decoupling (AC distribution will result in heavier weight distribution in the wings due to back-to-back converters at EPUs on the wings). • System may not be de-energised during interconnection, AC system will require synchronisation prior to connection.

Table 3 – Summary of influence of SDLs on EPS architecture design for case study aircraft.

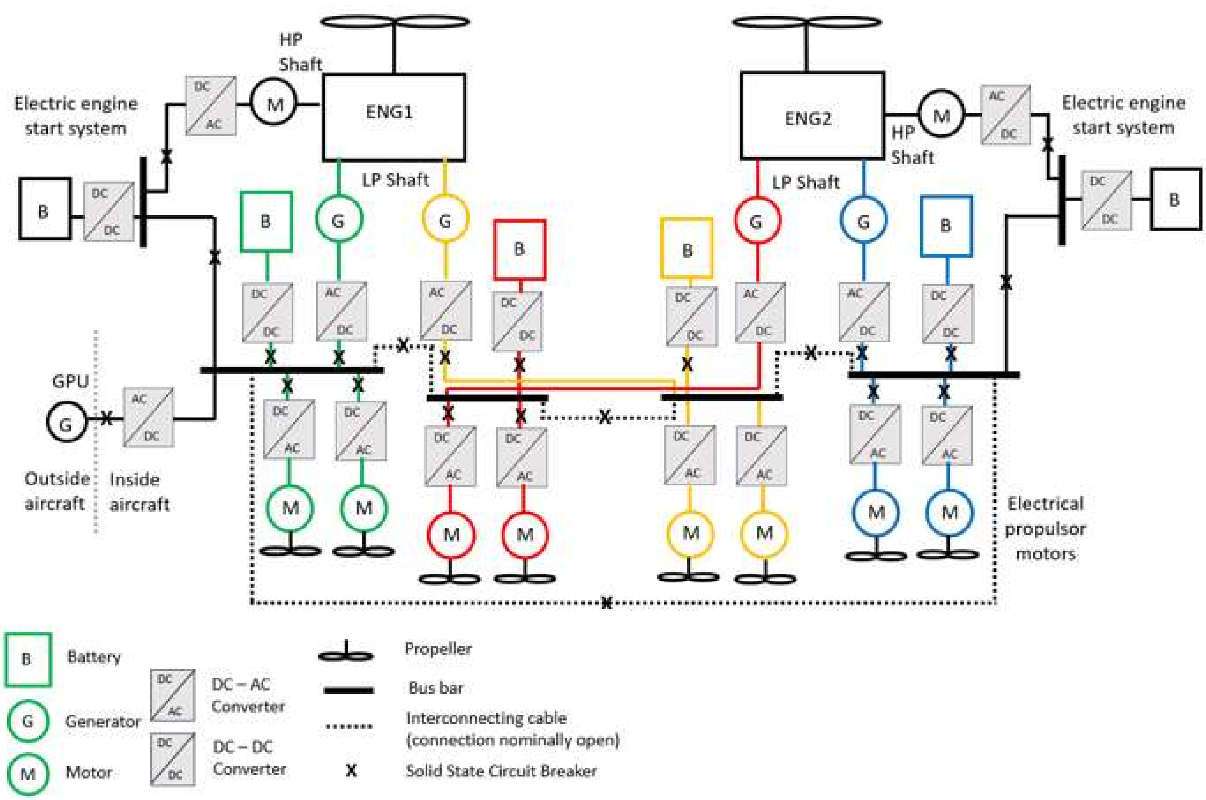

Figure 5: Derived baseline EPS architecture

Isolation between generators can be achieved by providing each generator and battery with a bus, resulting in a 4 channel system. Isolation provided by a 4 channel system is demonstrated by both the more-electric Boeing 787 [31], and the hybrid-electric IMOTHEP “Short-Medium Range Conservative” aircraft EPS designs [32][33]. A further FMS consequence of these decisions is that a zonal protection scheme is implemented, which enables the isolation of electrical faults to prevent propagation and enable healthy sections of network to continue to operate. It is proposed that resettable protection devices are used, to enable reconnection of equipment to the system after a fault has cleared.

For consideration of Power Availability, if power is to be maintained to the EPUs in the event of the loss of power sources, then interconnection between channels must be provided. Interconnection is also necessary for mid-flight electric engine start, and for engine start on the ground using a GPU. Power can be supplied to the starter motor via a battery (in the event of a dual engine failure) or generator from the other engine. Subsequently the second engine can be started electrically using electrical power generated by the other engine.

Under normal operation, channels operate isolated from each other. This subsequently leads to a choice of DC power distribution, to remove the need for frequency synchronization between AC distribution channels and for electrical decoupling of the speed of EPU motors and generators. A consequence of this decision is that if the GPU supply is 115 VAC, then an interfacing power electronic converter is required on the aircraft, to interface the GPU to the DC distribution system on the aircraft. The integration and design of the wider EPS for secondary, electrical power loads not directly part of the electrical propulsion drive train is outwith the scope of this paper.

From these decisions, an initial architecture shown in Figure 5 is derived. This is a 4 channel system with interconnection between channels, to enable electric engine start and supply electrical power in the event of a loss of engine. At this stage, the operational mode investigated is such that the electrical machine on the LP shaft is operated only as a generator.

3.2.3 Selection of Systems Design Trades

An initial failure rate analysis of the EPS was carried out to determine bottlenecks for the proposed EPS. From this it was determined that the main failure rate of concern, was that of 2 generators. Hence the initial system design trade to be investigated is the approach to maintaining sufficient propulsive power during an OEI failure. The calculated failure rate of 2 EPUs is higher (~1 .8 x 10^{-5} failures per flight hour) than that of 2 generator units (~1.6 x 10^{-6} failures per flight hour), but due to the higher numbers of EPUs, the loss of two generators was considered to be more critical.

Three approaches to redundancy in propulsive power in the case of an OEI condition were considered, as identified in row 1 of Table 2:

1. Adapt the hybridization factor such that power split between e-propellers and the turbo-prop is adapted to increase the proportion of propulsive power from the turbo-prop to mitigate the reduction in propulsive power from the e-propellers.
2. Maintain hybridization split between the turbo-prop and e-propellers and oversize generators by 100 % to mitigate the loss of the 2 other generators.
3. Maintain hybridization split between the turbo-prop and e-propellers, and oversize the batteries to mitigate the loss of 2 generators.

The design corner cases considered are normal operation, loss of two EPUs (motor and associated motor drive), and loss of 2 generators. The system must be sized for this failure to occur during maximum power (take-off and climb) and the aircraft must be able to perform a go-around in the event of a missed approach. In all three of these cases, the motors for the e-propellers are sized to accommodate the loss of 2 propulsor motors.

Datasets used for sizing in the Strathclyde Systems Sizing Tool for this case study are provided in Table 4. The aircraft is scheduled for entry into service in 2035, hence projected values for equipment performance up to this date were considered. There are a wide range of projected performance values available for equipment, the values selected below were chosen as a starting point. All equipment considered is for a conventional, non-cryogenic EPS. Studies to investigate the influence of different projected values were outwith the scope of this paper. The studies currently do not incorporate battery power density as a constraint.

EPS Component	Failure rate (failure rate/flight hour)	Power, energy or current density.	Efficiency (%)	Projected year
Motor	9.24×10^{-5}[23]	13 kW/kg [24]	96 [16]	2026
AC-DC power converter	2.7×10^{-5} [23]	22 kW/kg [24]	99[16]	2026
DC-DC power converter	2.7×10^{-5} [23]	15 kW/kg [24]	99 [16]	2026
Generator	1.3×10^{-4} [23]	21.3 kW/kg [25]	98.5 [25]	2035
Battery	9.31×10^{-5} [23]	400 WH/kg (pack level) [26]	98 [29]	2035
Solid State Circuit Breaker	1.7×10^{-7} [2l]	200 kW/kg [16]	99.5 [16]	2035
Cable	2×10^{-5}	170 A/(kg/m) [16]	99.9 [16]	2035
Busbar	1×10^{-6} [28]	Calculate based on copper density	99	-

Table 4: Datasets used in the Strathclyde Systems Sizing Tool for the case study.

4. Results for System Design Trades for Mitigation of OEI Scenario

Fig. 6 compares the weight of the EPS for the three different approaches to maintaining sufficient propulsive power to maintain flight in the event of an OEI scenario, for the four combinations of maximum and minimum (90% and 10 %) electrical power from the generators below the 900 m threshold, with maximum and minimum (90% and 10 %) hybridization factor. The power density of the EPS is 1.33 kW/kg for 90% hybridization factor and 90% electrical power from battery below 900m, and sizing the battery to provide additional power in the event of an OEI scenario. This is comparable to power density published for the EPS for a similar sized aircraft, of 1.0 -1.5 kW/kg [24]. The values published in [24] use power and energy densities for entry into service in 2026, and include a fuel cell, similarity in values obtained indicate the results presented here are acceptable.

Fig. 7 shows a sweep of EPS weight between these limits for electrical power split below 900 m and hybridization. The focus of the studies at this stage is to establish the design of the electrical section of the propulsion drive train, hence the results presented in this paper do not include electric engine start equipment, nor the EPS for non-propulsive systems. By inspection of Fig. 7, the EPS

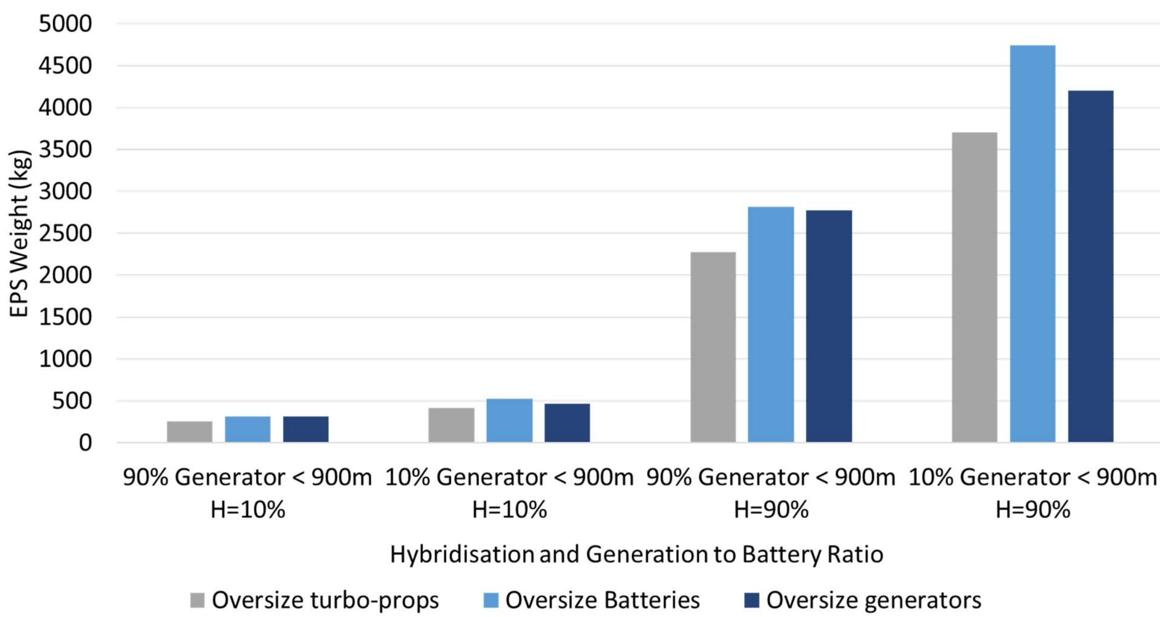

Figure 6: comparison of total weight of EPS with different levels of hybridization, *H*, power split between battery and generator below 900 m, and approaches to mitigate loss of a gas turbine and two EPUs, and with no oversizing to accommodate loss of EPUs.

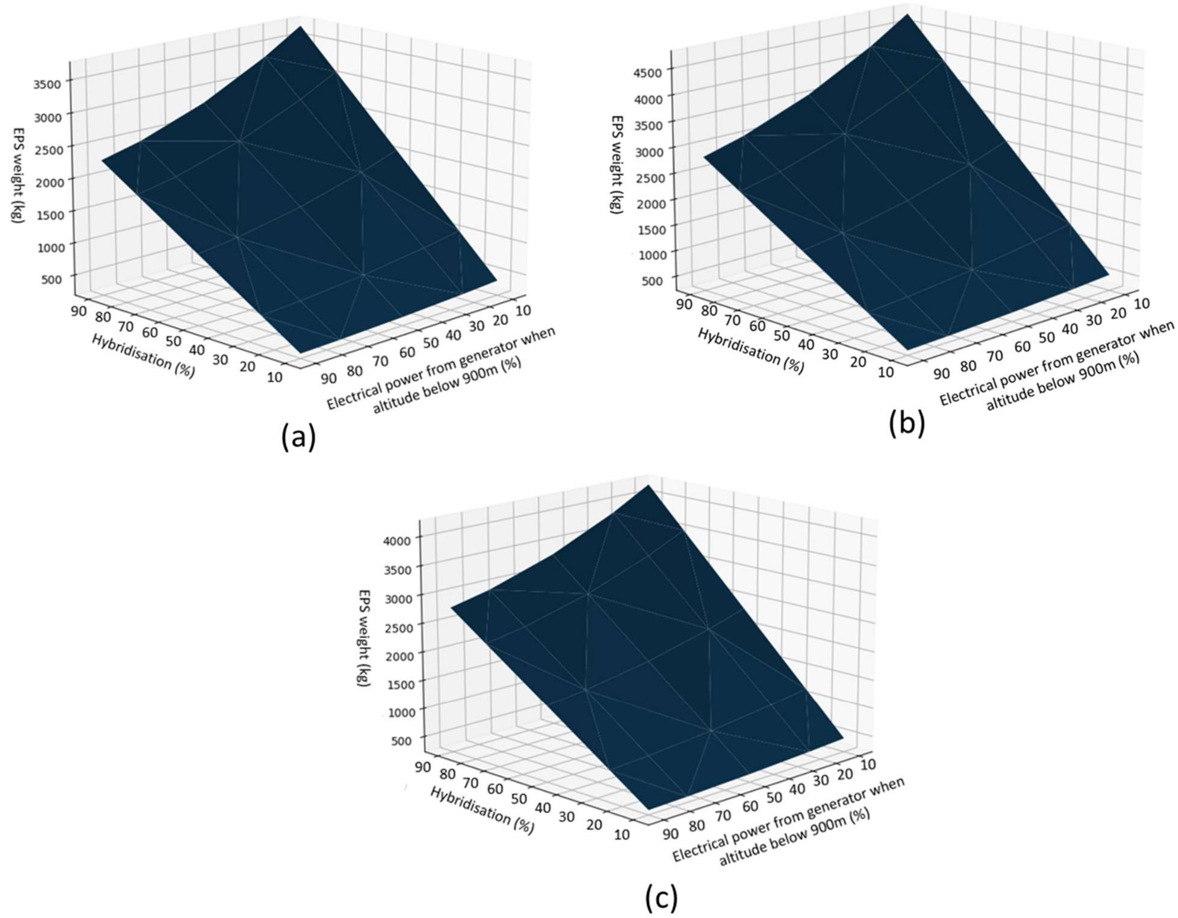

Figure 7: Variation of total weight with *H*, and variation of power split between generator and battery below 900 m, for oversizing turbo-prop (a), oversizing generators (b) and oversizing battery (c).

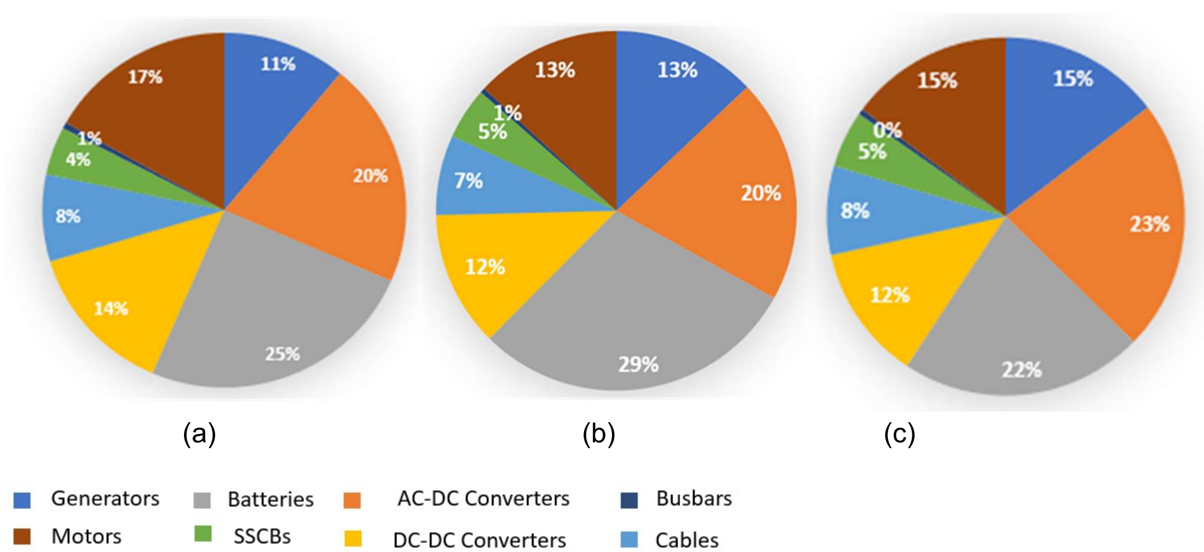

Figure 8: Comparison of weight breakdown (by percentage) for the case of 90% electrical power from battery below 900m for mitigation by oversizing turbo-props (a), oversizing batteries (b) and oversizing generators (c).

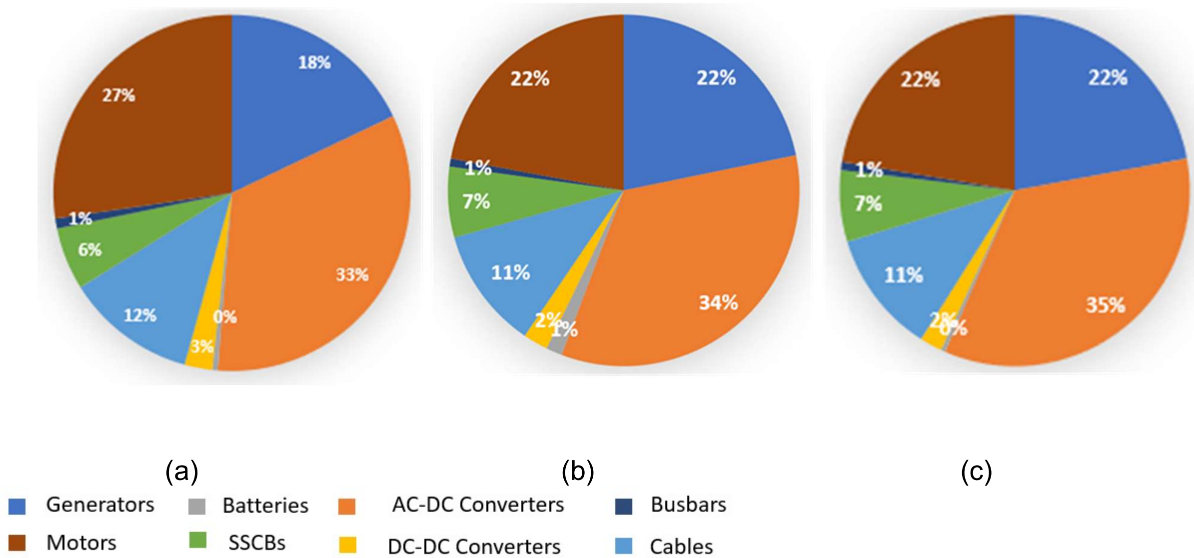

Figure 9: Comparison of weight breakdown (by percentage) for the case of 10% electrical power from generator below 900m for mitigation by oversizing turbo-prop (a), oversizing batteries (b) and oversizing generator (c).

weight is less sensitive to the power split between generator and battery below 900m, than the level of hybridization between the electrical and non-electrical propulsion systems. From the mission profile in Fig. 4, this is due to maximum power demand above the 900 m threshold. The generator must be sized to meet this maximum power demand, therefore generator size is sensitive only to the level of hybridization. Battery size is sensitive to both level of hybridization and electrical power split with the generator.

Figs. 8 and 9 show a breakdown of the contribution to EPS weight by percentage for the three different approaches to mitigate the OEI failure. For percentages under 0.5%, these are rounded down to 0%. Total weight of the EPS is sensitive to variation in hybridization but the percentage contribution from each type of electrical equipment remains constant as hybridisation varies. The contribution of the battery to total EPS weight very low when 90 % power is provided by the generator. For all approaches, electrical machines and power electronics are consistently significant contributors to EPS weight.

5. Discussions

5.1 Implications for Wider Propulsion Drive Train and Aircraft Design.

The MTOW of the ATR72 is ~ 22,000 kg, with a maximum payload of ~ 7000 kg. Therefore, with high levels of hybridization, the EPS for the aircraft propulsion system is 20 – 12 % of MTOW, depending on approach to mitigation of an OEI scenario and electrical power split between generator and battery below 900 m. From results presented, the approach of using the turbo-prop to provide additional propulsive power in the case of an OEI situation appears most attractive. However, it must be highlighted that the weights calculated in this paper do not include the additional fuel required for increase in electrical power from the generators (driven by the gas turbine), or increased fuel and impact on gas turbine and turbo-prop sizing in cases where the turbo-prop is required to provide additional propulsive power to mitigate the OEI scenario.

There is a direct conflict between the use of a battery to provide electrical power to minimize greenhouse gas emissions, particular at low altitude, and the weight penalty it attracts. Results presented in Section 4 all include sizing of the battery to accommodate a missed approach and go around scenario. Fig. 10 shows the impact on EPS weight if the approach is taken to size the battery to provide additional power in case of OEI, but has energy capacity for only one take-off, and not include the capacity for a missed approach and go-around. This is provided by an increase in power from the turbo-prop.

A major challenge for reduction of emissions at low altitude, is that maximum propulsive power is required when the aircraft is at these altitudes (take-off and climb). The conflict between EPS weight, power provided by the battery and emissions at low altitudes, illustrates the need for an integrated design approach with the wider propulsion drive train, to understand the impact that the use of a battery has on emissions, and fully optimize the aircraft design. This includes defining the upper threshold for the weight budget for the electrical propulsion system, to in turn guide the limits for hybridization factor and battery to generator ratio.

Further studies are needed to investigate approaches to optimizing the role of the battery in the EPS, to add value to the case for inclusion of the battery to reduce emissions at lower altitudes. These include using the battery to provide support to the EPS, and the limits of what support the battery can provide, recharging the battery during flight to provide support during descend and land, or electric taxi capability.

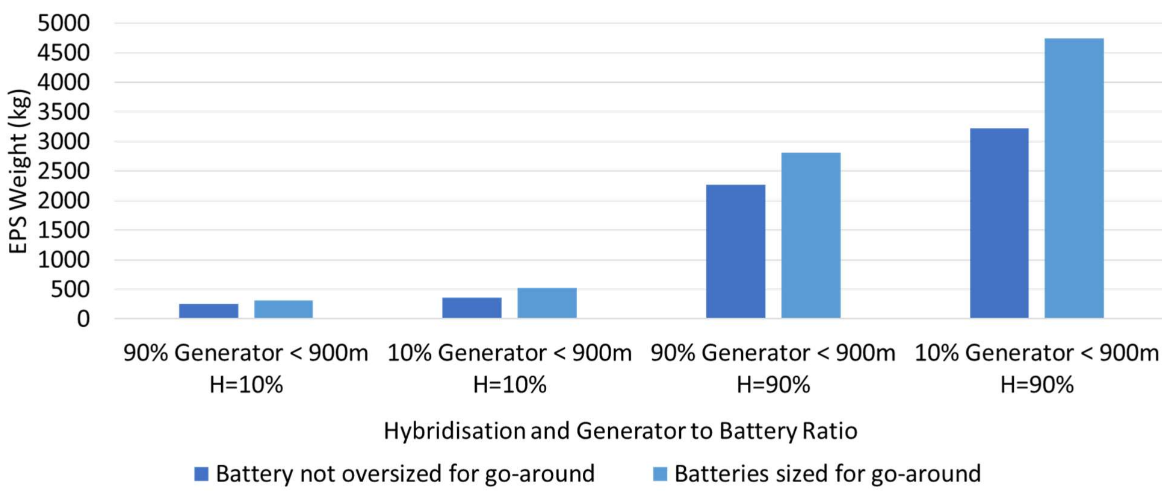

Figure 10 – comparison of sizing battery to provide additional power to mitigate OEI condition, where battery is sized only for one take off (not sized for go-around) and sized for normal take-off and climb and a missed approach and go-around condition.

5.2 Re-evaluation of System Design Levers

The SDLs of FMS in Section 3, following on from assumptions and consequences set out in Table 2, required that resettable protection devices be used to enable isolation of electrical faults. The need for interconnection, set out in Table 2 to enable electric engine start, and approaches to provide power to EPUs in the event of an OEI scenario, drives the SDL of Power Availability to ensure that any electrical source can supply any load. This subsequently leads to the approach to the SDL of Power Distribution to be selection of DC distribution. This enables interconnection between channels without the need for synchronization prior to interconnection: interconnection may need to take place between feeder channels where it cannot be assumed that one of the channels has lost power.

This approach necessitates resettable DC protection devices, which are an immature technology and availability is uncertain [30]. If availability of resettable protection devices within required timeframes are not considered possible, then alternative options must be considered and assumptions and consequences in Table 2 and SDLs revised. Alternative options may be the use of non-resettable DC protection, such as pyrofuses, or an AC distribution system. Electrical decoupling between generators and motors is still possible with AC distribution, but will require a back-to-back converter at each EPU on the wing, moving weight from the fuselage to the wings. The approach to interconnection, the need for, and approach to, sychnronisation prior to interconnecting channels assessed and the feasibility of performing this interconnection in response to a failure mode during take-off and climb (maximum power) investigated. For example, capturing the speed with which channel interconnection take place for sufficient propulsive power for the aircraft to maintain climb. This further indicates the need for interdependent design with non-electrical systems.

A further observation is the interdependency between EPS design assumptions and consequences (box “4”, Fig. 2) for different aspects of EPS design, which impact on SDLs. For example interconnection for Availability of Power, is necessitated for both assumptions for supply of electrical power to EPUs, and for electric engine start (via a GPU) and re-start mid-flight. However, synchronization is less likely to be of a concern for interconnection for electric engine start, as these interconnections would only be required either following the loss of a generator or feeder channel, or during the power-up phase (engine start using GPU).

6. Conclusion

A methodology to capture the baseline conditions for EPS design for a new concept DHEP aircraft, where very limited information is available has been presented. The methodology is driven by safety requirements of the aircraft, deriving from this the baseline failure conditions and ensuring that flight can be maintained. Using this approach to design a baseline EPS for a DHEP concept aircraft has demonstrated the need for the EPS to be designed in tandem with both the wider propulsion drive train (gas turbine, turbo-prop and e-propellers), and the wider aircraft design for determination of baseline requirements, including range of hybridization factor, mission profile. This tandem approach to design is critical for determining the optimum approach to mitigating failure cases, while minimizing emissions at low altitude. It will facilitate investigation of alternative response to failure conditions, including shedding of electrical loads to reduce the hybridization factor in response to a failure. The tandem approach is also needed to determine the acceptable weight, volume, losses (thermal management requirements) of the EPS, and system design trades with the wider propulsion drive train and aircraft design.

The approach presented enables the interdependency of different aspects of EPS design to mitigate failure modes, and their subsequent impact on SDLs to be captured, and combined in the EPS design, such that the resulting design is able to meet all failure modes identified. For example, the need for Availability of Power drives a need for interconnection between channels, required both for electric engine re-start and an OEI condition.

The case study has highlighted the need for careful consideration of availability of technology within the proposed timeline for aircraft development and entry into service. The impact of technology readiness may be very influential on SDLs. In particular resettable DC protection devices have been identified as a key technology if DC distribution is to be implemented.

Finally, the methodology does not include the incorporation of the EPS which is not directly part of the propulsive drive train. For example the EPS used on state of the art aircraft for more-electric engine functionality, and more-electric aircraft functionality including the environmental control system and electrically driven actuators. For a more-electric aircraft, this adds ~ 1.4 MW to the power rating of the EPS, adding 20% to the electrical power rating of the EPS for the concept aircraft in this paper. Hence further work to incorporate this secondary EPS into the full EPS design methodology is needed.

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9. Copyright Statement

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