Framework and Model Development for Aircraft Systems Integration in Conceptual-level Multidisciplinary Design Analysis

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Abstract

Framework and Model Development for Aircraft Systems Integration in Conceptual-level Multidisciplinary Design Analysis

Vijesh Mohan

The aviation industry has set ambitious goals to reduce its environmental impact significantly. To achieve these targets, innovative aircraft concepts such as hybrid-electric aircraft must be studied hand in hand with newer systems technologies to analyze the feasibility and improve efficiency. Typically, the impact of aircraft systems is quantified through empirical methods, that do not extend to novel systems technologies. Physics-based methods must be considered to evaluate the impact of these novel technologies. A focus on smaller commuter and regional aircraft is required as these are the categories that will first benefit from hybridization and electrification at current technology levels. Furthermore, in aircraft conceptual design, aircraft systems sizing and analysis need to be integrated into a multidisciplinary analysis and optimization environment to study the aircraft-level trade-offs and performance impact of the systems. This thesis presents a Multidisciplinary Analysis (MDA) framework that integrates systems sizing with other aircraft-level disciplines and can analyze the effects of novel systems architecture at the aircraft level. The framework uses physics-based system sizing methods and can analyze multiple architecture options such as more electric, all-electric, and unconventional systems architectures. System sizing studies of conventional aircraft that are performed using the presented framework are within an error range of 9-13%, which is acceptable for conceptual design. The framework also provides a robust platform for the integration of typically stand-alone analyses such as safety and thermal analysis. The presented MDA framework can also be used to study hybrid-electric aircraft configurations and integrate relevant disciplines such as fuel-system and energy storage assessment. A case study on a hybridized retrofit of a Dornier -228 aircraft is used to demonstrate the capabilities of this framework. Overall, the MDA framework presented in this thesis demonstrates that it is capable of doing an “integrated study” of novel aircraft and systems architecture and facilitates the integration of further disciplinary analysis, thereby improving the effectiveness of conceptual design studies and potentially improving confidence in novel aircraft and system concepts.

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Nomenclature

AC	Alternating Current
ACE	Actuator Control Electronics
AEA	All Electric Aircraft
ATA	Air Transport Association
BWB	Blended Wing Body
CAD	Computer Aided Design
CPACS	Common Parametric Aircraft Configuration Scheme
DC	Direct Current
ECS	Environment Control System
EDG	Engine Driven Generator
EDP	Engine Driven Pump
EHA	Electro-Hydrostatic Actuator
EMA	Electro Mechanical Actuator
EMP	Electric Motor Pump
FbW	Fly by Wire
FCC	Flight Control Computer
HMA	Hydro Mechanical Actuator
IPS	Ice Protection System
MDA	Multidisciplinary Analysis
MDAO	Multidisciplinary Analysis and Optimization
MDO	Multidisciplinary Optimization
MEA	More Electric Aircraft
MTOW	Maximum Takeoff Weight
OWE	Operational Weight Empty
PCE	Power Control Electronics
PCS	Power Consuming System
PGS	Power Generation System

PTDS	Power Transformation and Distribution System
PTU	Power Transfer Unit
RAT	Ram Air Turbine
THS	Trimmable Horizontal Stabilizer
TLAR	Top-Level Aircraft Requirements
XSDM	eXtended Design Structure Matrix

1 Introduction

The aviation industry is one of the fastest-growing industries due to ever-increasing transportation needs. International agreements and protocols are addressing the effort to improve aircraft efficiency to reduce the environmental footprints and fuel burn [1]. The industry considers newer aircraft configurations and novel aircraft systems technologies to achieve sustainable aviation [2]. Computer-based modelling techniques are used to explore and size the integrated aircraft systems architecture in newer complex aircraft configurations, which can handle the complexity more efficiently. The chapter provides the background and motivation behind the research work performed within the thesis.

1.1 Background

An aircraft is one of the most complex systems designed and manufactured for maximum efficiency. The design process involves several complexities not limited to aircraft configuration, systems architecture, and safety, making the product's development timeframe up to 10-20 years. The process can be time-consuming for novel aircraft configurations, such as hybrid electric and blended wing configurations, requiring complex aircraft systems architecture distinct from those currently available. For example, hybrid electric aircraft requires additional batteries and blended wing configuration requires novel flight control system, galley, furnishing, electrical system wiring distribution architecture. The aircraft design complexity, not limited to systems architecture and safety trade-offs for the configurations and technology, requires enhanced computer-based tools and frameworks enabling efficient ways of handling the complexity.

The aircraft design carried out today uses concurrent engineering process that interacts with multiple disciplines, such as design and manufacturing process, considering the entire aircraft product life cycle. The development process goes through several stages, including concept development, configuration selection and refinement, detailed validation, fabrication, and assembly phases. The industry adopts stringent validation processes for newer aircraft configurations. The details of aircraft development process can be found in [3–5].

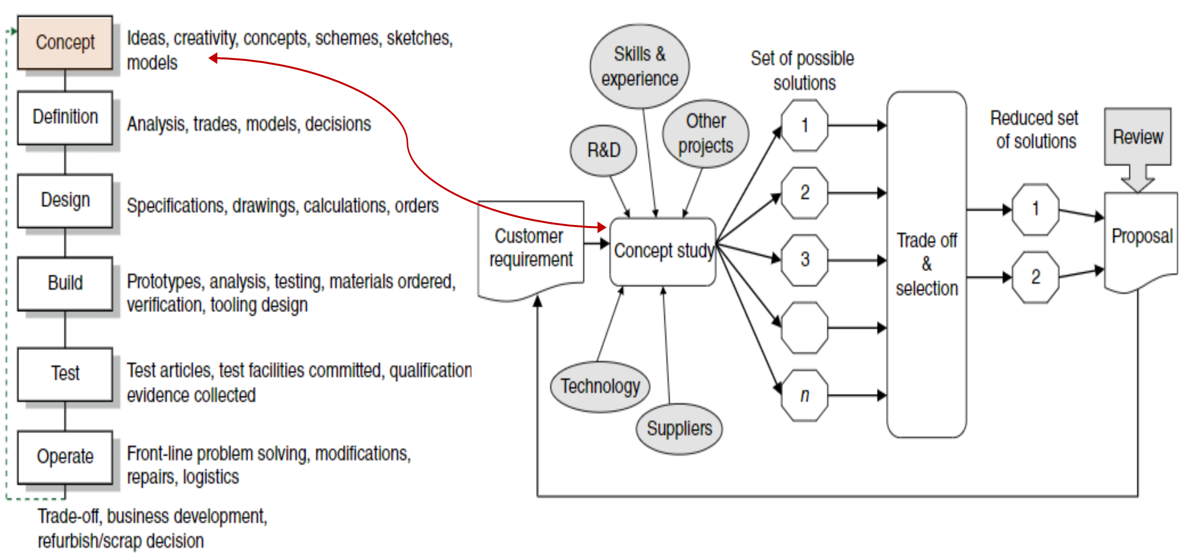


Figure 1-1 Aircraft Development Lifecycle [5]

The conceptual design phase remains where aircraft-level changes can be less costly and efficient. Figure 1-1 shows the aircraft development lifecycle focusing on the conceptual design. The key aircraft design phases are conceptual, preliminary, and detailed design. The conceptual design phase is the starting point of the aircraft design process. The design expert evolves aircraft-level requirements, compiling the customer requirements, technology availability, and supplier data sheets. The stage considers aircraft-level requirements, aerodynamics, engine requirements, performance, and structural and systems-level requirements to finalize the most promising aircraft configuration and technology for further design stage. The preliminary design stage adds the details of structures and systems. This phase includes several mock model development and wind tunnel testing. The detailed design phase involves finalizing and fabricating the actual aircraft model. Moreover, it involves multiple flight simulations to test the functionality of the design. Progressing through each design phase requires time and engineering investment, and redesigns can incur substantial financial losses. The current thesis focuses on and contributes to the conceptual design stage.

The existing methods are considered low-fidelity and are not validated to capture aircraft systems architecture variation. The methods use historically available data to size the aircraft and systems based on correlation. These are much quicker and more efficient ways to estimate the aircraft configurations, such as lengthening the fuselage to fit more passengers when the configuration and systems architecture remain conventional and unchanged. However, the methods are inconsistent for an entirely newer systems technology or configuration, which varies significantly from the baseline aircraft. Hence, analyzing an optimized configuration requires sophisticated methods.

1.2 Motivation

Aircraft systems are one of the disciplines responsible for providing safe aircraft functionality. These consist of subsystems or components working hand in hand to fulfill system needs. The ATA 100 chapters provide a brief overview of the systematic categorization of aircraft subsystems [6]. The categorization, not limited to aircraft systems, provides specific sections for disciplines such as maintenance and airworthiness, making interface identification easier. The aircraft systems are considered between ATA chapters 20-49. Appendix Table 5-8 provides a brief overview of multiple aircraft systems.

The aircraft systems design becomes more complex due to its interactions with other subsystems. Figure 1-2 shows the interaction of integrated subsystems in an aircraft. Although the aircraft systems operate independently, several subsystems are interdependent, such as flight control, which depends on the electrical system, and the electrical system, which depends on the propulsion system. The interaction becomes challenging due to the increased complexity of the availability of vast technology and dependency on disciplines such as safety, thermal, and maintenance requirements. For example, the avionics system depends on the ECS System for heat rejection and the electrical system for power consumption, which ultimately depends on engine fuel requirements.

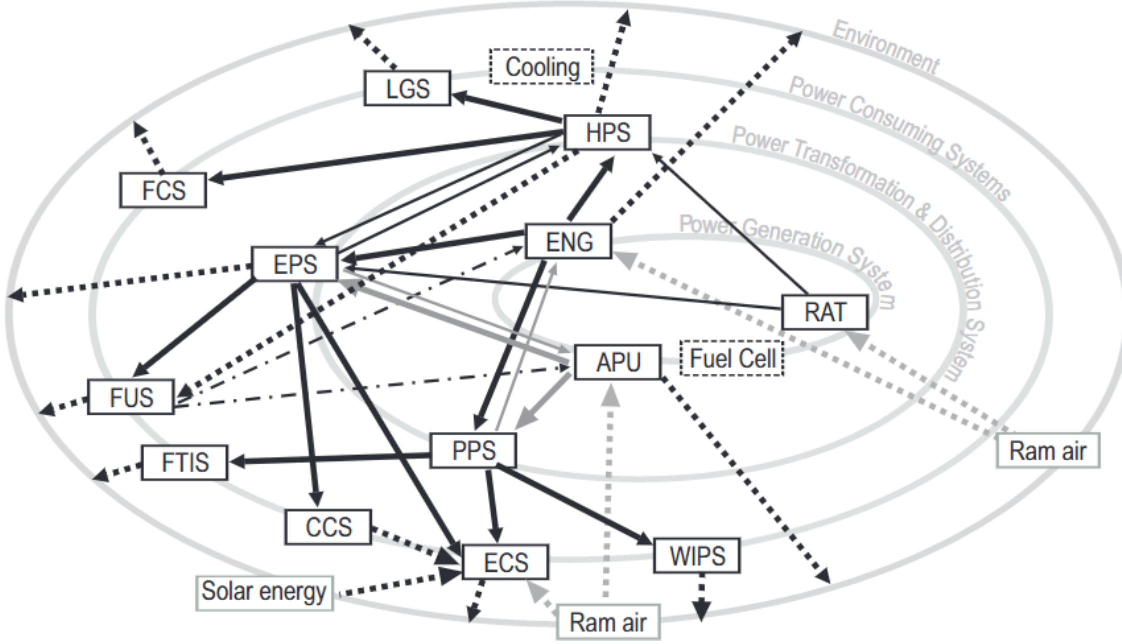


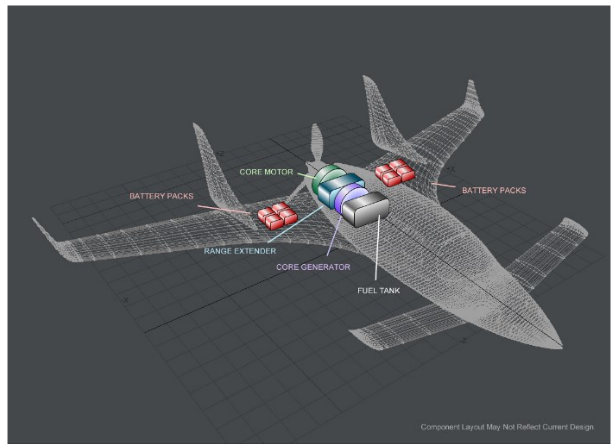
Figure 1-2 Interaction of subsystems [7]

Conventional aircraft systems refer to the basic subsystems and components installed in most traditional aircraft. In contrast, the More Electric Aircraft (MEA) subsystems refer to a shift in aircraft technology, increasing the use of electric power compared to traditional subsystems. Conventional aircraft consists of Engine Driven Pumps (EDP) of hydraulic systems and Engine Driven Generators (EDG) of electrical systems attached to the engine drive shaft to extract secondary power and then supply this power to the consumers. In a conventional aircraft, the emergency power unit includes an Auxiliary Power Unit (APU), Ram Air Turbines (RAT), and batteries. In an electrified aircraft, the classification depends on the systems architecture considerations. The overall integration allowed extraction of bleed air supply from the engine to the ECS and Anti ice system. The supply is used by the aircraft environment, hydraulic system, and fuel tank pressurization[8]. The flight control system in a conventional aircraft interacts with the hydraulic system for the required amount of hydraulic flow to fulfill the actuation needs of the aircraft. The hydraulic system depends on the engines, which request fuel from the fuel system to satisfy the requirements. Eliminating the bleed air off-take and the hydraulic power subsystems reduces the engine power off-take. Boeing 787 incorporated dedicated ram air inlet technology to eliminate the engine bleed air offtake. Bleedless ECS architecture fuel-saving benefits are significant [9].

The use of conventional aircraft systems employed in aircraft such as Boeing 737 and Bombardier Challenger 300 contribute heavily towards carbon footprints. The MEA and AEA architecture increased the number of electrical system users, interacting extensively to fulfill the functions [4]. The FCS and bleed-less ECS systems are electric and designed for a higher voltage level. The critical technology variations increased the overall load on the electrical system. The broad number of architectures available to be sized requires sufficient complex tools that will model and enable interaction between other disciplines.

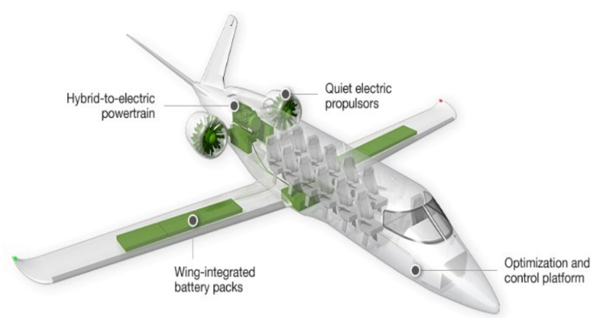
Unconventional aircraft, such as hybrid and distributed electric architectures, have more connected subsystems than conventional and MEA. Figure 1-3 shows some of the recent hybrid electric aircraft configurations. The existing methods do not capture the highly complex interaction between electrified propulsion and electrical systems. The conventional aircraft propulsion system consists of a combustion subsystem, dependent on the fuel system for the required fuel to generate thrust. The engine feed system typically transports the fuel stored in the wing. In contrast, Hybrid and distributed electric aircraft consist of dedicated batteries and electric motors to fulfill the propulsion energy needs. Brelje et al. [10] extensively review multiple hybrid electric architectures. The overall hybridization and electrification will aim to carry less fuel in the aircraft. An integrated environment is required to study the aircraft systems architecture variations with respect to the adapted hybridization factor. The adapted MEA and AEA architecture must align with the actual electrical energy capacity of the current or targeted aircraft timeline. The energy density of the current battery technology needs to be increased to fulfill the entire electric aircraft energy requirements. Hence, these are typically applied to the dedicated flight phase [11]. However, the energy requirements of aircraft systems can also be met with improved battery technology in the future [12]. Unconventional aircraft types such as Blended Wing Body (BWB), Box-Wing, and morphing wings are studied to have high fuel-saving characteristics [13].

a) Faradair bioelectric hybrid electric [14]



b) Volta Volare DaVinci [15]

c) Heart ES30 hybrid electric [16]



d) Zunum hybrid electric [17]

Figure 1-3 Hybrid electric aircraft

Apart from the weight, yet another area of focus is the space consumed by aircraft systems. The overall battery energy density of current batteries is much lower. The NASA X-57 battery energy density at the cell level is about 225 Wh/kg, whereas the requirement is about 400 Wh/kg [18]. The space consumed by the battery in terms of volumetric energy density is seen to be directly proportional to the battery energy density. Hence, improvements in battery energy density are expected to bring improvements in volumetric energy density. Since, the current battery technology has low battery specific energy density, the batteries will consume more space than fuel. The space allocation for the batteries is relevant for the propulsion systems and additional batteries, which might lead to the relocation of aircraft systems. The scenario is also interdependent on the overall safety in terms of redundancy and maintainability of subsystems. For example, a battery susceptible to thermal runaway requires thermal risk assessment integration. The thermal risk assessment integration within the workflow will enable qualitative analysis of aircraft systems in terms of temperature rise, which can be used to allocate additional cooling subsystems or relocation of subsystems. The relocation of aircraft systems due to the allocation of batteries might require access panels for maintenance, requires integration of maintainability tool. A sophisticated integration platform is required to analyze existing electrical subsystems, such as batteries, that don't meet the minimum safety standards requirements [19]. The allocation and relocation of aircraft systems based on the engine rotor burst zone, which considers space exposed to engine burst debris [20]. A dedicated integrated environment must be developed to carry out a detailed analysis of the interaction of the integrated aircraft systems with the electric propulsion architecture.

The aircraft systems considerations in conventional aircraft design methods are limited to a small set of architectures. The low-fidelity methods use statistical correlations to estimate the aircraft systems contribution toward the aircraft level. Sometimes, these are limited to a single set of configurations or do not consider systems-level parameters. Moreover, the vast interactions between the subsystems are not well incorporated within the design stage. Many possible architectures in terms of technology are not explored along with other novel disciplines such as maintenance, safety and thermal due to their interaction. The aircraft design is driven by the business needs. The optimal design may have several trade-offs for design, manufacturing and maintainability. Direct operating cost (DOC) is one of the aircraft design's main parameters. The aircraft systems significantly contribute about 30% to the aircraft level parameters such as Operational Weight Empty (OWE), DOC, Direct Maintenance Cost (DMC) and Aircraft Development Cost [21]. Moreover, fuel consumption and drag are other key aircraft-level parameters which are highly influential [9]. One essential method for the overall optimization of the aircraft is to enable a high-fidelity systems architecture design capability. The design needs to be studied hand in hand with the interaction of integrated systems at the aircraft level within conceptual design.

1.3 Thesis Scope & Objectives

The objective of the thesis is to build a sandbox multidisciplinary integration framework to analyze the impact of aircraft systems sizing effects at the aircraft level. Figure 1-4 shows Concordia University Aircraft Systems Lab's proposed MDA framework. The framework is designed to create an aircraft Multidisciplinary Analysis (MDA) platform, with a key emphasis on exploring the integration of various aircraft systems. The primary objective of this platform is to facilitate the incorporation of novel disciplines such as safety and thermal considerations alongside aircraft systems. The goal is to completely examine interactions between them and effects at the aircraft level. The sandbox MDA framework developed in the thesis is built as a part of the work. The main objective is the development of the capability to integrate novel subsystems analyses, such as the Aircraft System Safety Assessment Tool (ASSESS) [22,23] or the thermal risk assessment [24] or Solar Power System (SPS) and to analyze the impact within the MDA environment. The possibility of integrating novel disciplines is explored by connecting tool inputs and outputs, such as solar power systems, to analyze the impact at the aircraft level.

Yet another objective of the thesis is the development of detailed aircraft systems sizing models. The thesis presents a methodology to conduct the aircraft systems sizing and its interaction in a multidisciplinary environment in the overall conceptual aircraft design and covers the capability to conduct system architecture trade-off studies. The overall scope is limited to the aircraft systems sizing and integration studies with multiple disciplines such as aircraft sizing, energy storage, fuel system, and other systems for commuter and regional aircraft categories and the unconventional aircraft configurations limited to hybrid electric configurations since the retrofitting and redesigning of these aircraft are potential paths toward achieving hybrid and all-electric flight, which are even more challenging objectives from a system integration perspective. The application is limited to a few of the chosen critical aircraft systems, such as flight control, hydraulic, and electrical systems, as these subsystems contribute heavily towards electrification. Moreover, the technology variations studies can be conducted easily by considering these subsystems. The methodology also considers all other systems, such as galleys, entertainment, avionics, etc., in terms of simplified estimations to conduct overall systems estimation effect. To determine the energy needs and weight variations at the aircraft level, conceptual designers need to study the aircraft electrification strategy hand in hand with the propulsion systems at the system and subsystem levels to determine the overall feasibility of a hybrid-electric aircraft concept. Additionally, the proprietors of smaller and commuter aircraft need to consider the potential impact of changing the certification category (i.e., from Part 23 to Part 25) due to a weight increase caused by the weight of the additional batteries.

Another objective of the thesis is to demonstrate the possibility of integrating large set of disciplines allowing seamless interaction between disciplines. The capability is attained in terms of connectivity within a remote based platform which allows it to connect with novel expert disciplines at various organization.

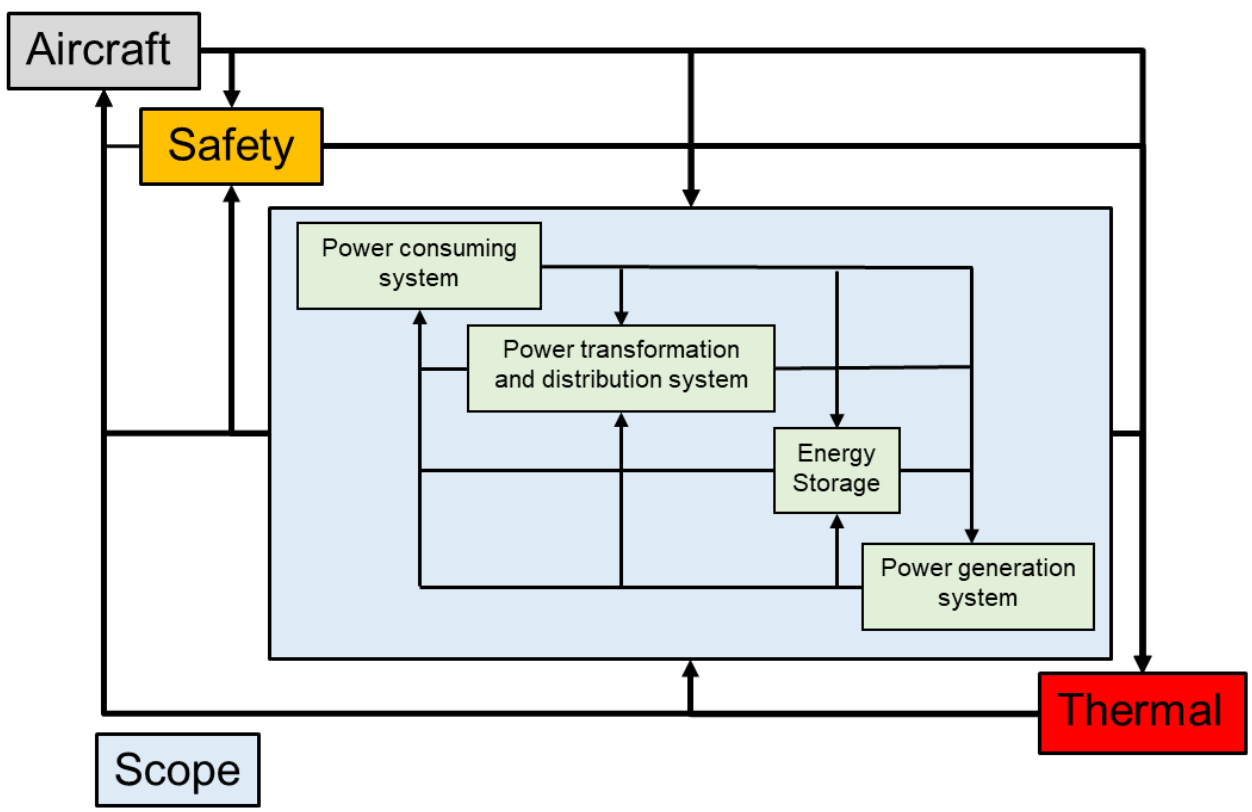


Figure 1-4 Proposed multidisciplinary analysis (MDA) framework.

To summarize, the following research objectives are addressed: -

1. Develop an MDA "sandbox" to test novel (sub-)system integration methods, such as safety analysis, thermal analysis, or maintenance.
2. Develop detailed aircraft system sizing models and integrate systems sizing, novel disciplines into MDA workflow.
3. Demonstrate system architecture trade studies for more-electric and hybrid-electric aircraft within a multidisciplinary design analysis (MDA) environment.
4. Demonstrate the possibility of integrating large set of disciplines within the MDA framework.

1.4 Organization of Thesis

The thesis is structured as follows. Chapter 2 provides the state-of-the-art aircraft systems sizing and integration works. Chapter 2 also identifies the need for a framework to analyze the aircraft systems. Chapter 3 provides details on the Multidisciplinary Analysis framework to study the systems effects at the aircraft level. Chapter 3 also provides detailed system sizing methods to integrate in the MDA environment. Chapter 4 shows the application of the framework with several case studies on hybrid electric aircraft. Chapter 5 provides concluding remarks and the scope of the future work.

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2 State of the Art

This chapter provides the state-of-the-art aircraft systems sizing and integration works. It also identifies the need for a framework to analyze the aircraft systems.

2.1 Aircraft Systems Integration Frameworks in Early Aircraft Design

The conceptual design of the aircraft evolves based on the customer needs, which in turn is translated through estimations of overall weight, performance, and cost. The initial considerations of the systems evaluation were presented by Roksam [25], Torenbeek [26], Raymer [27] and Howe [28] during the early 1960s. The aircraft system contributes to overall system weight toward the aircraft's Maximum Take-off Weight (MTOW). These methods are primarily derived from historically available data and experience. Statistical curve fittings methods such as power laws, polynomial and linear relations are used to correlate the system weight to the aircraft level parameter.

The conventional methods focus on the variations of system weight due to the change in aircraft configurations. Figure 2-1 shows aircraft systems considerations in the early design framework. The application of the methods varies between the aircraft based on the validation cases (for example, whether similar existing aircraft is already validated) and the aircraft type. The aircraft manufacturers would typically consider specific methods based on the required aircraft configuration. For example, the Cessna method is established explicitly for general aviation aircraft weighing 12000 lbs and lower [25]. General Dynamics (GD) method for transport and business jet aircraft weighing more than 12000 lbs [25]. Torenbeek and Raymer's method covers almost most of the conventional aircraft configurations. [25]

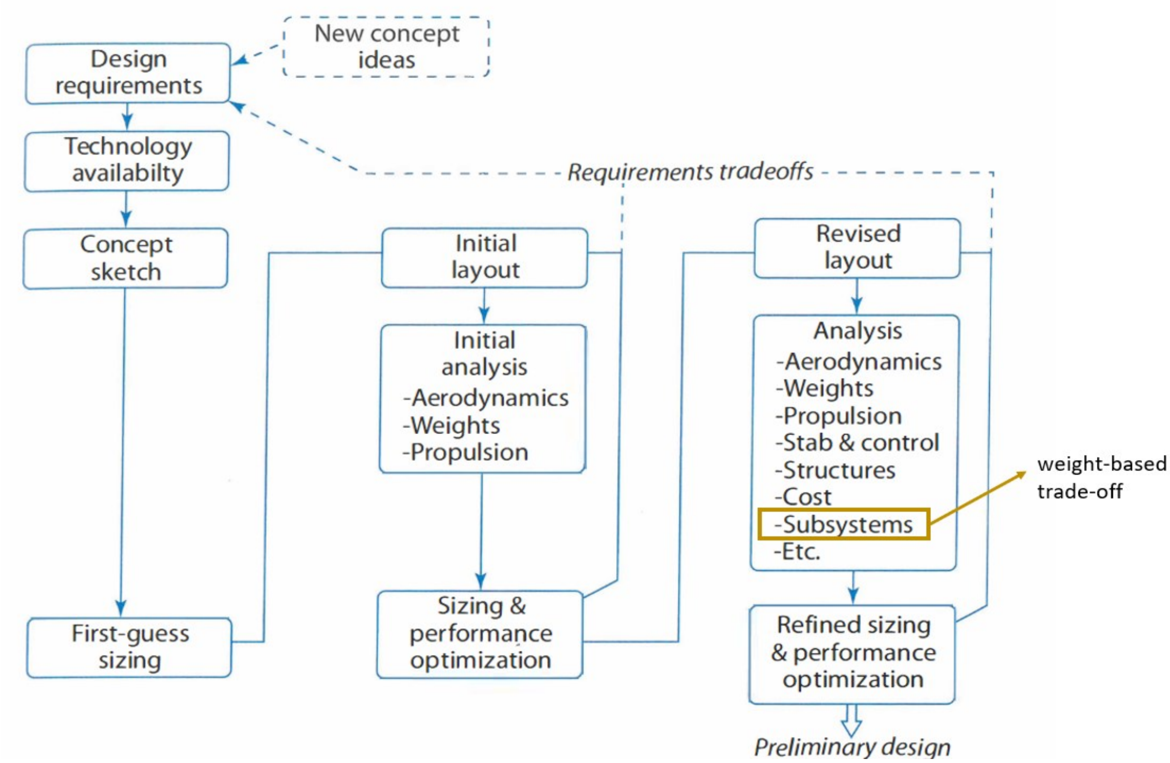


Figure 2-1 Systems consideration in early aircraft design framework [29]

Moreover, the conventional methods are still not valid for unconventional aircraft configurations such as Blended Wing Body (BWB) or distributed propulsion. For example, the cabin volume or wing area estimation for these configurations remains unconventional.

Table 2-1 compares the traditional system weight estimation methods. The conventional methods use subsystem weight build-up to communicate with aircraft-level parameter MTOW. During the conceptual design stage, the design expert can choose from various methods with varying levels of parameter complexity, depending on the application's requirements and data availability. The Howe method adopts a linear approach to systems weight interaction with the MTOW of the aircraft. Torenbeek and (General Dynamics) GD method consider additional parameters, such as cabin length and volume, for specific subsystems, such as the Environmental Control System (ECS) and the Ice Protection System (IPS). The Raymer and NASA methods consider subsystem weight analysis without using MTOW. Most relevant aircraft-level parameters are correlated to estimate the system weight. For example, the Flight Control System (FCS) weight estimation uses control surface area.

The methods such as Howe, Torenbeek, GD, and Raymer are well suited when the basic assumption on the aircraft remains conventional and the systems architecture remains the same. However, when there is a significant deviation in the aircraft configuration, the historical data are not established to cover these cases. Another drawback of the methods is the grouping of the systems together. For example, the Torenbeek method groups ECS and IPS system weight estimation, which makes applying different subsystem technology to a specific subsystem difficult. Moreover, the conventional methods are still not valid for unconventional aircraft configurations such as Blended Wing Body (BWB) or distributed propulsion. For example, the cabin volume or wing area estimation for these configurations remains unconventional.

Table 2-1 Early aircraft design Systems Integration methods

Method	Year Established	Parameter Complexity	General Trend	Aircraft Types	Accuracy (Conv.)
Torenbeek [26]	1982	Low	Polynomial	Ga, T, B	High
Howe [28]	2000	Low	Linear	Ga, T, B	Low
GD [25]	1985	Low	Polynomial	T, B	Low
Nasa FLOPS [30]	2018	Medium	Linear	Ga, T, B, BWB	High
Raymer [27]	1992	High	Polynomial	Ga, T, B	Low

Abbreviations: Ga – General Aviation, T- Commercial Transport, B – Business Jet, BWB - Blended wing body

The NASA Flops method presented in reference [30] considers systems weight empirical estimation to include additional unconventional configurations and technology considerations to differentiate the subsystems by including additional aircraft-level parameters. For example, the fuel system weight estimation considers the number of engines to cover distributed propulsion architecture, and the hydraulic system weight estimation considers the hydraulic system pressure, wing variable sweep weight penalty, fuselage planform area, number of engines to account for distributed propulsion. Horvath et al. [31] have critically reviewed the NASA method, comparing a case study with conventional B737 aircraft. The study concluded that the FLOPS method which is very much like the existing methods captures the overall component weight trends. Moreover, the method captures some of the aspects to quantify unconventional configurations such as blended wing body and distributed propulsion aircraft.

The conventional methods are limited to aircraft system weight estimation. However, the trade-off analysis must consider additional parameters, such as power and fuel consumption, to meet the current aviation industry emission targets. Yet another drawback is the difficulty of quantifying novel subsystems. For example, it is not possible to study the contribution of electrical distribution within the electrical system by varying the technology parameter. Although the NASA method considers technology parameters, novel subsystem technology additions require additional effort. Due to this reason, the complete system representation might be challenging to achieve with the traditional empirical methods. The objective must be to conduct a trade-off analysis considering all available or novel technologies to evaluate aircraft-level feasibility.

2.2 Towards Physics-based Systems Integration frameworks

To address the shortcomings of the methods presented in the previous section, several researchers have developed so-called “physics-based” methods and also expanded the scope of system considerations beyond weight estimation. Physics-based methods involve creating representative mathematical models and equations that represent the physical processes or subsystems and then using computational tools to solve these equations and make predictions about the functioning of the subsystem. The physics-based modelling begins with identifying the subsystem, functions and their dependencies with other subsystems. As a next step, detailed mathematical models are built to represent the subsystem functions and interactions. The simulation of the models is then possible by connecting subsystems to mock the real-world application. The physics-based methods use subsystem integration at the aircraft level, enabling the additional capability of the contribution of subsystems.

Koeppen [32] was the first to apply the physics-based method, introducing functional systems architecture models. Individual systems models are built to estimate the weight and power consumption based on mission. The overall power consumption can be used to estimate the secondary power off-take regarding hydraulic flow demand. The inputs remain aircraft-level inputs with sufficient scaling factors of the aircraft type under the analysis. One downside of the models is that the unavailability of the factors could lead to high errors, or this is required to be quantified. Another drawback of the method is that it cannot be adapted to include novel systems technology. This requires additional model building and calibration of existing models. Although models exist for power consumption at the generation level, the unavailability of subsystem power consumption can prevent the possibility of integrating innovative systems.

Liscouët-Hanke [7] carries out integration based on the subsystem power flow sequence. Figure 2-2 shows the subsystem dependencies. Power Consuming Systems (PCS) are the subsystems which consume secondary power. For example, FCS, ECS, etc., require power depending on the flight phase. The subsystems responsible for fulfilling the power requirements of PCS by transforming the input power are categorized under the Power Transformation and Distribution Systems (PTDS) category. The electrical system, hydraulic system, and pneumatic system are the systems considered within PTDS. Power Generation System (PGS) provides power to the PTDS systems. The Engines, Auxiliary Power Units (APU), and Fuel Cells are the subsystems considered within the PGS.

The methodology allowed trade studies between multiple MEA architectures. The major parameter interface consists of aircraft-level inputs and subsystem-level technological and functional parameters. As a first, the researcher considered additional subsystem parameters such as power-off take, drag and fuel consumption for aircraft-level evaluation of MEA architectures. The methodology is applied to Airbus aircraft, focusing on power systems. As a downside, the study focuses on minimal aircraft configurations and safety parameters for electrical systems. Additional effort or methodology expansion is required to analyze smaller hybrid electric aircraft.

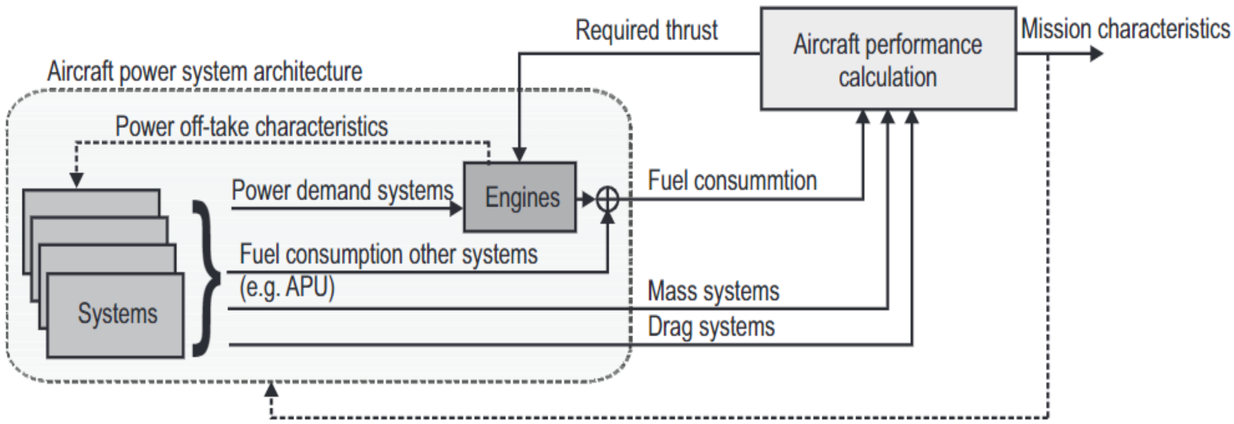


Figure 2-2 Systems Integration framework using energy flow method [7]

Lammering [33] adapts a physics-based method to integrate subsystem parameters such as mass, centre of gravity (c.g) and power off-take. The aircraft-level integration allows multiple systems architecture trade-off analysis within the conceptual design stage. The iterative design approach between aircraft design and systems sizing resizes the aircraft and captures the snowball effect. His work focuses on exploring the possibility of introducing innovative systems modelling. Probabilistic methods to quantify the uncertainties at the aircraft level include identification and downselection of technology interdependency, parameter association with the selected dependency and allocation of probability distribution to the parameters.

Chakraborty [34] adapts the methodology to describe detailed models for FCS, IPS, ECS, Pneumatic Systems, Hydraulic Systems, Electrical Systems, landing gear systems, electric taxi systems, and Thrust Reverser actuation subsystems. The methodology applies an automatic architecture definition algorithm called “Candidate Subsystem Architecture Descriptor” as a key contribution. The research explores the large design space for commercial aircraft, encompassing MEA and AEA configurations. The automatic architecture descriptor allows subsystem architecture analysis without requiring analyst input or Subject Matter Expert (SME).

Table 2-2 compares conventional, recent empirical and physics-based methods for FCS sizing. The subsystem-level quantification capability of the physics-based method can be extended towards many architectures, which can then interact with other disciplines, including safety, thermal and maintenance.

However, the modelling and methodology scope is limited to commercial aircraft. Hence, the architectural variations for smaller commuter aircraft, part 23 types, require additional modelling effort.

Table 2-2 Comparison of conventional, recent and Physics based methods for FCS sizing.

Method / ***Attribute***	***Torenbeek***	***NASA Flops***	***Liscouët-Hanke***
MTOW	✓		
Wing Area	✓	✓	✓
Control Surface Area	✓	✓	✓
Mechanical Actuation	✓	✓	✓
Electric Actuation			✓
Power off Take			✓
Subsystem quantification			✓
Large Architecture			✓

2.3 Systems Integration within Multidisciplinary Analysis (MDA) Framework

The aircraft systems consideration in a conventional aircraft design primarily based on weight. In this approach, interactions with other disciplines are not necessarily considered, as there is no exchange of inputs or outputs between the disciplines. Multidisciplinary Analysis (MDA) allows the interactions between disciplines, allowing more integrated trade-off analysis. The approach enables early identification of complications and allows informed decision-making regarding trade-offs. Using MDA to evaluate system performance yields comparatively more comprehensive results at the aircraft level.

The top-level Multidisciplinary Optimization (MDO) framework involves optimizing the aircraft with performance and cost. Although this process does not include optimizing individual components, it allows multiple trade-off analyses and interactions depending on the connecting disciplines and complexity. The main objective of the process is to ensure that the final design meets the cost and performance objectives meeting the Top-Level Aircraft Requirements (TLAR).

As an early effort, Raymer [35] implemented the optimization algorithm and built the automated capability of resizing the aircraft as per changes in the design attributes. Yet another top-level MDO framework is Preliminary Aircraft Design and Optimization (PrADO), developed by Westphal et al. [36]. PraADO consists of several dedicated modules for multiple disciplines. It is possible to carry out design and trade-off analyses at the end of convergence. The modular structuring enables the possibility of conducting MDO for unconventional aircraft configurations. NASA developed an open-source MDO framework, OpenMDAO [37–39]. One of the key features of OpenMDAO is the easy integration capability of different design disciplines.

The early MDO framework includes weight-based trade-off and optimization and uses empirical systems correlations. These methods lack detailed aircraft systems aspects within the optimization framework. For example, it is not possible to analyze fuel savings due to the electrification of subsystems, as the primary focus is on the weights, which may lead the optimizer to a non-conclusive minimum. Hence, the necessity of integrating aircraft systems within the top-level MDO frameworks enables a more comprehensive understanding of different systems' interactions.

More recent MDAO frameworks (e.g., AGILE 4.0) include aircraft systems with more details within the framework, enabling trade-off analysis between aircraft systems at top-level aircraft analysis. Lammering [33] integrates aircraft systems models within the Multidisciplinary Integrated Conceptual Aircraft Design and Optimization (MICADO) framework. The MICADO framework consists of specific modules to cover disciplines. XML file called Aircraft Exchange (AiX) carries out the input-output propagation, ensuring the independence of disciplines and enabling the capability of carrying out standalone analysis.

Chiesa et al. [40] established the methodology to integrate aircraft systems within the ASTRID (Aircraft on board Systems Sizing and Trade-off Analysis in the Initial Design phase) tool. The systems-specific ASTRID methodology mainly focuses on mass and power estimation [41]. ASTRID tool is applied within the Top-level European collaborative framework AGILE 4.0 [42]. Fioriti et al. [43] use the framework to showcase the framework capability to evaluate electrified small transport aircraft. The case study considers AEA architecture for on board aircraft systems. The impact is shown at aircraft-level parameters such as MTOW and SFC. However, the tool is not available as an open source. Junemann et al. [44] integrate aircraft systems architecture, sizing and assessment within AdVanced Aircraft CONfiguration (AVCON) project. AVCON is a collaborative conceptual design specifically for mid-range aircraft. The aircraft systems design methodology corresponds to *GenSys,* enabled in collaboration with Hamburg University of Technology (TUHH). The overall aircraft level analysis capability alongside SysFuel+ enables a mission simulation tool capable of resizing the aircraft.

Table 2-3 Capability comparison of aircraft systems sizing methods

	Attribute / ***Method***	***Weight***	***Power***	***Drag***	***Unconventional / More Electric***	***System Architecture***	***MDAO Integration***	***Commercial Aircraft***	***Commuter Aircraft***	
									Part 23	***Part 25***
Early Methods	Howe	✓						✓	✓	✓
	Torenbeek	✓						✓	✓	✓
	Raymer	✓						✓	✓	✓
	Roksam	✓						✓	✓	✓
Recent Methods	NASA Flops	✓						✓	✓	✓
Physics Based	Koeppen	✓			✓	✓		✓		
	Liscouët-Hanke	✓	✓	✓	✓	✓		✓		
	Lammering	✓	✓		✓	✓	✓	✓		
	Chakraborty	✓	✓	✓	✓	✓		✓		
	Chiesa	✓	✓	✓	✓	✓	✓	✓	✓	✓

Table 2-3 presents an in-depth capability comparison between the methods. All the methods are capable of conducting weight-based trade-off analysis. However, the physics-based methods quantify additional aircraft-level parameters. More recent MDAO frameworks, such as (AGILE 4.0) incorporate physics-based methods, enabling more extensive design space exploration, developed parallel to the current framework.

The collaborative MDAO efforts by multiple authors carried out simplified estimation in quantifying the impact of systems and their technology. However, these methods still lack details regarding the availability of the required space for systems within the aircraft. As an early effort to incorporate the systems volume, Raymer [45] defines Net Design Volume (NDV) as a simplified estimation by comparing the total available volume to the total incurred volume. NDV represents the available volume for all components, such as structures, avionics systems, equipment, landing gear, routing, and access provisions.

Liscouët-Hanke and Huynh [46] further enhance the methodology to introduce Equivalent Design Volume (EDV) to estimate the required system space, including maintenance, ventilation and access. The required parameters are extracted from conceptual 3D models or output from the initial sizing of the aircraft. The component space requirement is estimated using a bottom-up approach using aircraft and systems-level parameters for small and business-type aircraft. Budinger [47] utilizes the scaling laws to calculate the dimensions and installation space requirement for the flight control actuation system components.

Tfaily et al. [48] further implemented a parametric tool called (Catia Advanced Design Linking and Iterations Software and Tool) CATALIST within the CATIAv5 platform. The method gives a preliminary estimation of the volume for system installation. The main interface is the excel spreadsheet capable of interacting with the MDAO environment, which uses Design tables and design rules to automate system component geometry generation. The simplified representative shapes used for components eliminated the detailed parameter requirement for components, simplifying the early design space estimation. As a downside, the methods are not yet implemented for MEA or All-Electric Aircraft (AEA) aircraft.

Sanchez et al. [24] proposed a framework to overcome this by introducing a CAD modeller within the MDAO workflow. The CAD modeller is split into external and internal aircraft geometry. The external aircraft geometry is a result of the aircraft sizing results and allows the propagation of basic aircraft-level inputs to estimate systems and other disciplines. The internal CAD modeller is a resultant of the analysis used to propagate systems dimensions to the CAD modeller to generate internal aircraft geometry. The overall integration enables the interaction between safety, thermal and maintenance criteria based on real-time systems placements, which is crucial for aircraft such as hybrid electric, where the trade-off analysis between battery and fuel tank space claims are required.

2.4 Summary: Need for an Integrated methodology.

The chapter reviews the existing approaches of systems consideration in conceptual aircraft design. The conventional methods are limited to aircraft system weight estimation. Further considerations of parameters such as power and fuel consumption are not possible with these methods. Moreover, these methods are not expanded to quantify subsystems. Hence, it is difficult to study the contribution of subsystems at the aircraft level. It is required to consider novel subsystem technology to conduct a large qualitative analysis.

The recent Physics-based methods showcase the capability to represent subsystem architecture. Furthermore, it is possible to include novel subsystems and analyze their effects at the aircraft level. The method is available across multiple literatures, but several notable challenges exist. First, there's a limitation in the availability of detailed mathematical models for subsystems based on publicly available data. Consequently, there's a compelling need to expand and construct detailed subsystem models for academic purposes. Furthermore, existing methods lack applicability and validation for smaller aircraft, particularly for commuter and regional unconventional aircraft types such as hybrid electric configurations. Hence, there is a crucial requirement to develop and validate a set of models tailored to this specific application.

When considering Multidisciplinary Analysis and Optimization (MDAO) frameworks for system integration, existing frameworks focus on weight, power, and cost interactions. However, to enhance the scope and flexibility of these frameworks, there is a need to establish a new framework. This framework should allow for more detailed system architecture activities encompassing safety, thermal considerations, maintenance, layout, and other critical factors. This expansion is essential to improve the model structure and facilitate more comprehensive architecture evaluations.

In summary, the existing literature provides a foundation for physics-based system methods, but there is a need to build detailed mathematical models, especially for academic use. Additionally, MDAO frameworks should be expanded to cover a broader range of system architecture activities, necessitating an enhanced model structure for novel subsystem connectivity and in evaluating complex design aspects.

3 Methodology: Aircraft Systems Multidisciplinary Analysis Framework

This chapter provides details on the Multidisciplinary Analysis framework to study the systems effects at the aircraft level. This chapter also covers detailed system sizing methods to integrate in the MDA environment.

3.1 Multidisciplinary Analysis Framework

Conventional conceptual aircraft design involves the interaction of multiple disciplines, such as structures, systems, aerodynamics, propulsion, etc., addressed separately based on specific independent design requirements for disciplines. There are often limited interactions between the disciplines. Due to this, choices large-scale trade-off analysis is often not available. Moreover, the subsystem analysis with multiple architectural possibilities is neglected. The aircraft Multidisciplinary Design and Analysis (MDA) follows an integrated approach, considering the interaction of multiple disciplines with top-level aircraft design. The design disciplines are often interconnected and interact in an iterative process. The subsystem consideration within an analysis framework involves considering multiple novel technology possibilities. The broader design space considerations can improve optimal design, enabling effective and efficient trade-off analysis. The integrated interaction increases the overall performance of the design process. For example, the impact of systems design variations is readily available to estimate the new aerodynamic characteristics such as drag. The process improves overall confidence level in the design point.

The multidisciplinary aircraft analysis framework is depicted in Figure 3-1 using the eXtended Design Structure Matrix (XDSM) [49]. XSDM allows a graphical representation, visualizing the data dependencies and process flow path between the disciplines. The framework integrates various modules such as *Aircraft sizing*, *Aircraft geometry*, *Energy Storage*, *Fuel System*, and *Systems MDA*. The *Aircraft sizing* module encompasses the hybrid electric and all-electric aircraft sizing capabilities. The basic structural evaluation is included within the empty weight correlation. The module consists of basic propulsion system architectures for conventional, hybrid electric and all-electric configurations with the possibility of expansion of novel propulsion top-level architectures. The subsystem sizing methods are integrated within the *Systems MDA* module. The *Systems MDA* discipline encompasses the aircraft systems and subsystems sizing and performance estimation modules. The *Systems MDA* allows for evaluating the impact of aircraft system architectures on aircraft-level parameters such as MTOW and fuel burn. Finally, the *Systems MDA* module drives system architecture and decision-making activities. Figure 3-1 presents the overview of the MDA framework for the systems integration studies for hybrid electric aircraft. The framework presented in this thesis have been disseminated and shared through research papers [50,51]. The inclusive scope of this thesis encompasses the overall MDA framework, integration and interaction of each discipline with detailed sizing information for each tool provided in section 3.1.1referring to the contributing author.

The framework is compatible with conventional and unconventional small commuter and regional aircraft types. The conventional type of aircraft is straightforward. The hybrid electric power train consists of additional challenges. The fuel system is the most impacted subsystem due to the adapted hybrid electric configuration. The hybrid electric configuration consists of batteries or fuel cells. The specific configuration, which involves placement of energy storage, which can include batteries or different fuel types within the wings, leads to less space requirement or availability for the fuel tanks. As stated in the previous section, a three-dimensional (3D) geometric or Computer-

Aided Design (CAD) modeller updates the available space assignment of subsystems within the aircraft design. The design expert can manipulate the modeller inputs to assign system-specific space and reassign it based on the priority of systems. The hybridization of the power and specific energy density of the battery is the major parameter involved in the sizing and resizing of the batteries and fuel tanks. The overall integration at the aircraft level ensures that the parametric assignments interact with subsystems and analyze the effects on aircraft-level parameters.

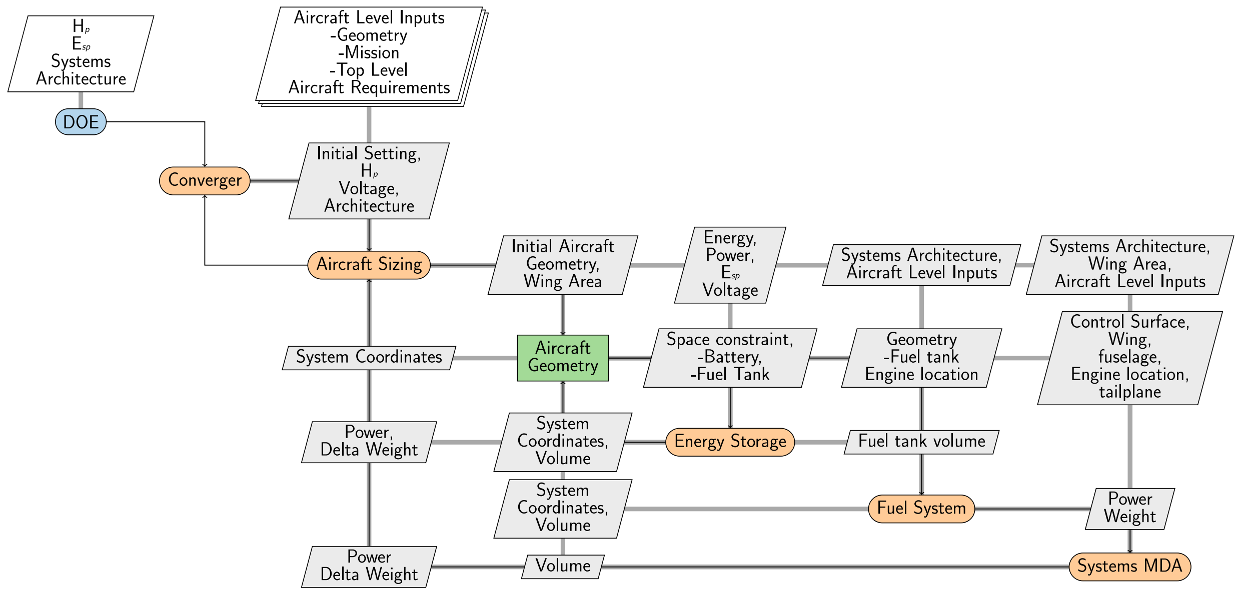


Figure 3-1 Overview of the multidisciplinary design analysis workflow for the system integration studies for hybrid-electric aircraft.

The major focus is given to the energy storage and fuel system modules, as the hybridization level of the aircraft directly affects the modules. The higher hybridization factor leads to less fuel requirements as major power requirements are met by the energy storage (batteries as considered in current architecture). The framework considers the separation of the modules to analyze the effects at the aircraft level and subsystems, which also allows investigation of the possibility of novel technologies such as hydrogen storage in addition to the batteries investigated in the current framework. The separation of the tools within the framework allows to drive the geometric arrangement due to new systems such as batteries. Ultimately, the method ensures the aircraft design is consistent with the latest propulsion and aircraft systems changes, resulting in a more efficient convergence.

The overall initialization of the integration process begins with the input generation. The major inputs are the Top-Level Aircraft Requirements (TLAR), which are user-defined. These customer requirements are the starting point of the analysis. The analysis iteration process must converge towards satisfying the TLAR or should look for different solutions to meet the needs. The subsystem level requirements are generally derived from the TLAR, such as technology choice limits, which can vary depending on the level of electrification the customer expects and depend on the specific aircraft entry timeframe, such as battery-specific energy density. Formalizing the design requirements into design inputs and variables is considered the first step of the design instance generation.

The multi-disciplinary analysis, one of the most complex processes due to the interaction of multiple disciplines, is required to enable efficient data transfer of design input variables, which requires standardization to reduce source error and improve the consistency of variables. The Common Parametric Aircraft Configuration Scheme (CPACS) [52] is an XML-based data format developed by the German Aerospace Center (DLR) to enable information exchange between design and analysis tools. The highly customizable structuring allows a single CPACS file to be updated by several tools and makes the updated outputs available for further analysis as input.

The current framework considers a single centralized data repository (CPACS). The centralized approach ensures that all the disciplines interact with single entity, allowing the data generated by one discipline to be readily available to any discipline. Moreover, the approach provides a structured way to track the design changes. Furthermore, different design teams working on various aspects of the aircraft can share and access data from a centralized CPACS repository, enabling large-scale discipline integration possibilities.

3.1.1 Multidisciplinary Interaction with disciplinary tools

As part of the presented work was to integrate several tools developed by other graduate students in the aircraft systems lab, a short description of these tools is presented, focusing on their interactions with the tools in the scope of this thesis.

Aircraft Sizing

The initial step within the design framework is to generate the relevant aircraft sizing, which can be used as an initial estimate for other disciplines. The current framework considers CLARITY tool for hybrid-electric aircraft sizing [53]. The tool is designed to carry out conventional propeller-based small commuter aircraft and modified parallel hybrid versions. The tool takes various inputs from the CPACS, such as aircraft design parameters, mission specification and TLAR for the weight, constraint diagram and mission analysis. The tool follows an iterative process to estimate the aircraft MTOW, fuel fractions and battery weight fractions over the mission for convergence.

The aircraft sizing framework is the first discipline in the design process that provides initial estimates of the MTOW, including the OWE of the aircraft. In the case of a hybrid electric configuration, the framework also provides initial estimates of the battery weight and Energy requirements for each flight phase. The fuel weight required to carry out the mission is updated within the CPACS file, which stores the inputs and outputs of several disciplines. In addition to the weight estimates, the initial aircraft sizing also provides basic geometry parameters such as wing area, which is used in further estimation by the aircraft geometry tool. Overall, the initial aircraft sizing and subsequent geometry estimation provide the foundational parameters for the rest of the design process.

The overall initialization of the tool begins with the initial estimates of the overall MTOW within the weight module. The tool uses initial estimates from empty and fuel weight to estimate the overall weight of the aircraft. The constraints are derived by carrying out performance objectives at various flight segments. The Hybridization of Power (H_p) and Battery Specific energy Density (Esp) are major interface parameters for multiple disciplines. The aircraft sizing framework defines the parameter H_p on a scale of zero to one, which means that, for a power value of zero, the power associated with the electric motor would be zero and vice versa. The framework is also designed to carry out Part 23 and Part 25 aircraft types.

Aircraft Geometry

The aircraft geometric modeller is typically responsible for creating and updating the aircraft's geometric parameters. The geometric modeller is used to interact with aircraft design experts and disciplines to extract or modify the physical shape and dimensions of the aircraft [24]. The detailed geometry generations are then made available for further analysis. The analysis results and the disciplinary geometry requirements are translated back to the modeller, creating an integrated design. Typically, in the aircraft design, geometric modellers are fed back to the GUI to visualize the procedure depending on the chosen aircraft parameters and results.

The proposed workflow integrates the geometric modeller to interact with *Aircraft Sizing*, *Energy Storage*, *Fuel System* and *Systems MDA*. The aircraft sizing tool provides initial design parameters such as wing area. Following the initial step discussed in the previous section, the workflow generates geometric parameters required for other disciplines. As the next step within the workflow, the design expert can manipulate the basic fuel tank layout as a space claim volume and allocate it to batteries or fuel tanks. This initial decision serves as a constraint for the energy storage tool, which will prefix the layout and can update the tool according to the discipline volume requirement. The cuboidal parameters of the batteries are used to compare the space allocation constraint to finalize the space requirement. As a simplified estimation of the overall volume, the three-dimensional parameters of the fuel tanks are estimated by dividing the fuel tanks into several subsections according to the wing layout for the volume estimation. The geometric parameters can be updated to finalize the fuel tank layout, which can be used by the physics-based fuel system tool. Based on the aircraft sizing results, the Systems MDA tool uses the geometric modeller to forward the updated wing area requirement. The geometric modeller further uses the wing area to manipulate the control surface area requirements. The updated aircraft level parameters are made available to analyze other physical systems.

Energy Storage

The energy storage module consists of subsystems capable of capturing and storing energy produced from different sources, which can be used upon demand. The used case includes energy to propel the entire aircraft to energy needs as small as to smooth out energy demands for a continuous grid supply. Some Energy Storage technologies include batteries, fuel tanks and hydrogen storage. The current framework investigates the integration of batteries and conventional fuel-based fuel tanks within the Energy Storage System. The Battery Sizing tool within the Energy Storage framework is adapted from [54] and integrated within the MDA framework.

As per Heit [53], the module considers inputs along with geometric and electrical constraints. The geometric constraints include the length, height and width in terms of available space. The integrator allocates the initial space assignments available from the aircraft geometry module to the energy storage module specific to the batteries and fuel tanks. The subsystem level inputs, basic cell level information such as type of cell, dimensions, voltage and capacity, are predefined and fixed during the process.

The integrator considers internal and external parameters to manipulate the integration process. The top-level aircraft parameters such as pack level dimensions, Maximum Voltage, and Energy and Power requirements are considered within the external parameters. These are considered alongside H_p and E_{sp} to drive the battery sizing workflow. However, the module level dimensions, cell level requirements and voltage requirements which are required for the tools, are considered within internal parameters. These are manually manipulated or internally predicted using empirical correlation concerning the constraints.

The battery sizing module provides outputs corresponding to the battery's final dimensions, energy, and capacity. The integrator updates the initial dimensions and sets with the final external dimensions obtained from the battery. The integrator then allocates the remaining available space back to the fuel tanks within the aircraft geometry tool, ensuring the required amount of fuel is available within the wings. If a higher hybridization factor is chosen, the initial battery allocation space can be increased to yield more space for the batteries. However, the current framework restricts the packing of the battery irregularly, ensuring the battery remains cuboidal and placed within certain limits from the centre of the aircraft to the engine location. The current framework only considers fuel tank modification for integration without considering the fuselage for battery placement. This approach also ensures that the placement of the system within the fuselage remains unaffected within the workflow.

Fuel System

The fuel system consists of subsystems responsible for supplying fuel to the propulsion system. The safety critical subsystem transfers the fuel to control the aircraft's (C.G). Hence, it consists of additional subsystems according to the architecture and fuel tank layout. The workflow considers an architecture-based fuel system tool developed by Rodriguez and Liscouët-Hanke [55]. The subsystem allocations within the fuel tank layout are used to estimate the volume and space requirement for the fuel system.

The primary interaction of the integrator with the fuel system tool comes as a subsequent step after finalizing the energy storage space claim. The total available space for the fuel system allocated by the aircraft geometry is considered an external parameter of interaction, which, along with other parameters such as architecture, technology and certification, are used to allocate the subsystems internally. This subsystem allocation is considered an internal interaction with respect to the workflow. The external parameter, such as fuel system weight and geometric layout, is updated in the workflow and available to the Systems MDA tool for further analysis.

3.2 Systems MDA Framework

The Systems MDA module comprises aircraft systems and subsystems sizing and performance estimation modules. The Systems MDA module is a primary module which interacts with the multiple disciplines to carry out system-level architecture evaluation and mission-level analysis to access the systems architecture impact at the aircraft level. The interaction of the Systems MDA module is primarily with the CPACS-based descriptor, which will be explained in the later sections. The section will explore the sizing aspects of the Systems MDA tool.

Figure 3-2 shows an overview of the Systems MDA module in an XDSM format. The Systems MDA XDSM structuring allows improved interdisciplinary communication, easy disintegration of disciplines. For example, separation of fuel system and energy storage is considered as these subsystems has highest impact in a hybrid electric configuration. The aircraft systems sizing within the Systems MDA is structured in three major subsystem categories as proposed by [7], power consuming systems, power transformation and distribution systems, and power generation systems. The categorizing of the subsystems is based on their functions and power flow patterns. The framework's fully nested systems analysis allows for sizing at each level.

The Systems MDA sizing process will initiate with the availability of aircraft level, updated geometric modeller parameters and the aircraft systems interdisciplinary inputs. The major interface parameters are the power and weight of the subsystems. The systems sizing shares the interface with a physics-based fuel system module developed by Rodriguez and Liscouët-Hanke [55] to obtain power and weight estimates. The energy storage sizing determines the stored energy required to operate secondary power systems and should not be confused with the energy storage sizing for propulsive power. The subsystem level integration with the physics-based method along within the aircraft sizing environment allows the systems architecture and trade-off analysis to analyze the effects at the aircraft level. The framework is built to adapt easily and include additional subsystems (e.g., a fuel cell or solar power system). Table 3-1 presents the considered subsystems and an overview of the associated modelling assumptions.

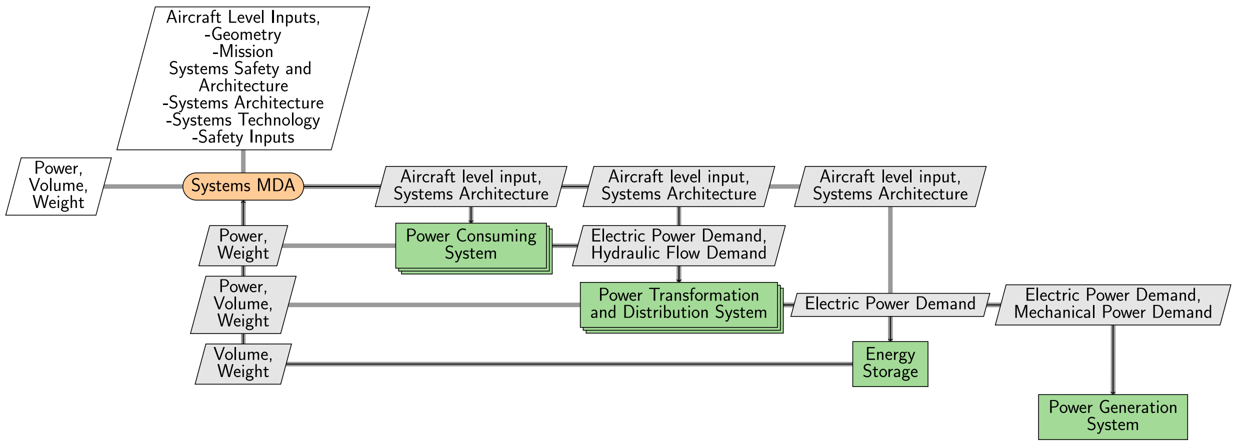


Figure 3-2 Overview of Systems MDA Integration

The power-consuming systems are the subsystems which interact directly to fulfill aircraft functionality and consume power for their operation. The subsystems include galley, furnishing, entertainment, flight control system, avionics, lighting, environmental control systems and ice protection system. The method of categorization can be found in [7]. As a first step of the sizing process, all the consumer subsystems are sized to determine three types of power demands and weights. The type of power demands is electric power demand, hydraulic flow demand and mechanical power demand. The power flow pattern allows the power-consuming system to accumulate the net power demand based on the consumers for further analysis. Table 3-1 shows the subsystems categorized within the power-consuming system, adapted modelling method and the systems architecture possibility. The flight control system is modelled using the physics-based method with subsystem-level integration. The galley, furnishing and entertainment are modelled using empirical correlation with aircraft-level parameters. All the other systems are modelled using the NASA flops method [30]. The power-consuming system is also built to include novel consumers to access the impact at the aircraft level.

The analysis follows the sizing of the power-consuming system to fulfil the power needs of the consumers. The power transformation and distribution systems are the subsystems which are not capable of generating their own power but fulfill the needs of consumers by converting the power from one form to another. Some subsystems fall into these categories: pneumatic, hydraulic, and electrical, and they are further classified based on their functionality into distribution, conversion, and generation. For instance, electrical power conversion includes transformers, inverters, and converters, electrical distribution encompasses electrical feeders, and electrical power generation encompasses generators. The first step in the sizing process is estimating power demand on distribution subsystems. The aircraft level parameters and systems architecture are used to estimate the losses incurred due to power transportation. This is then added to estimate the new power demand that the generation will fulfill. Additionally, the conversion losses are considered in terms of efficiency. Table 3-1 shows the subsystems categorized within the power transformation and distribution system, the adapted modelling method and the systems architecture possibility. The sizing details for the subsystems are explained in the later sections.

The sizing of the power generation system follows the power requirement analysis of the power transformation and distribution system. The power generation system consists of subsystems which can generate their own power. These subsystems fulfil the needs generated by the power transformation and distribution system. Subsystems such as Engines and APU are categorized under the power generation system. The sizing of the power generation requires the power requirement by the power transformation and distribution system and the system architecture of the subsystems. Typically, engines are the primary component responsible for fulfilling the secondary power demand needs. The mechanical power is extracted from the engine shaft to fulfil the needs of generators and hydraulic pumps. Additionally, smaller aircraft use APU and RAT for emergency power and ground power fulfilment. The power demand is used to size the subsystems. The power generation system is modelled in a way to be able to connect additional novel power generation subsystems such as solar power systems.

Table 3-1 Overview of implemented systems sizing tools, their sizing methodology, architecture or technology options, and level of granularity.

Systems MDA	**Subsystem**	**Size Estimation Methodology**	**System Architecture Type or Technology**				**Level of Granularity**
			Electrical	**Hydraulic**	**Pneumatic**	**Mechanical**	**Major Components**
Power-Consuming System	Flight Control System	Physics-based	✓	✓	n/a	✓	Major components
	Environment Control System	NASA FLOPS [30]	✓	n/a	✓		Complete system
	Ice Protection System	NASA FLOPS [30]	✓	n/a	✓		Complete system
	Galley, Furnishing, Lights, Entertainment	Empirical	✓	n/a	n/a	n/a	Complete system, per subsystem
	Avionics, Instruments	NASA FLOPS [30]	✓	n/a	n/a		Complete system
Power Transformation & Distribution System	Hydraulic System	Physics-based		✓	n/a	✓	Major components and length of piping
	Electrical System	Physics-based	✓		n/a	✓	Major components and length of wiring
Power Generation System	Auxiliary Power Unit (APU)	NASA FLOPS [30]	✓	n/a	n/a	✓	Major component

Due to the subsystems, the aircraft systems sizing analyses the effects of weight variations at the aircraft level. However, it is crucial to analyze the effects of subsystems on the environmental impact. One of the key targets of the aviation industry to reduce the environmental impact is fuel burn reduction, thereby reducing carbon emissions. The aircraft system impact on the fuel burn must be carried out within a specific mission profile to estimate the fuel burn contribution of subsystems. The Systems MDA mission level analysis capability will be elaborated in the later part of the section.

3.3 Detailed Systems Sizing Estimation

3.3.1 Flight Control Systems

The Flight Control System (FCS) is responsible for actuating control surfaces that allow the pilot to maneuver the aircraft. The FCS comprises all systems and subsystems except the Flight Control Computer (FCC), which signals the actuators and is considered part of the avionics system. This module can analyze traditional, mechanical actuation, more electric, and all-electric type actuation. This section will explore the methods used for detailed sizing of FCS systems and subsystems, the inputs/outputs to the module, such as aircraft level parameters, and the systems architecture.

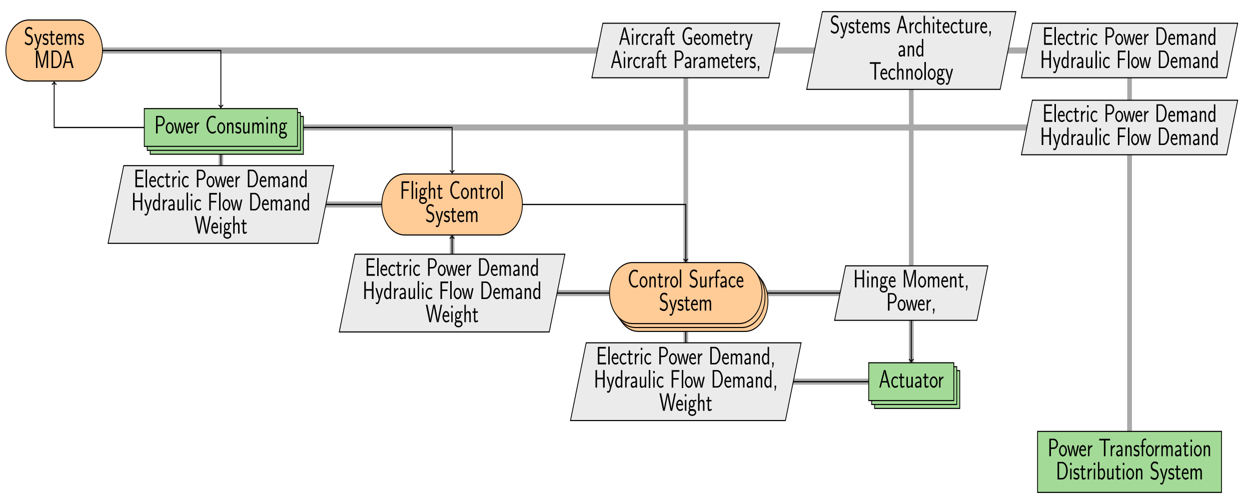


Figure 3-3 Flight Control System sizing workflow overview

The FCS sizing analysis consists of power-based sizing of subsystems, which are then used for further analysis of other connected systems, such as the Power Transformation and Distribution System (PTDS) and Power Generation System (PGS). Figure 3-3 shows the FCS sizing approach in an XSDM format. The Control Surface System. The analysis of individual subsystem is followed by integration into the Control Surface Systems which integrates all the associated subsystems within a control surface. This is further integrated into FCS module, allowing Power Consuming System for further analysis. Figure 3-4 shows the detailed FCS sizing flow chart. The initial step of the sizing process consists of estimating the hinge moment of the control surface. The hinge moment refers to the aerodynamic hinge torque acting on the movable control surfaces such as the aileron, elevators, and rudder of the aircraft. The control surface deflection leads to aerodynamic load due to the relative wind, which creates a moment around the hinge line. Hence, the power requirement of the actuators, depending on the architecture, is derived from the hinge moment of the control surface.

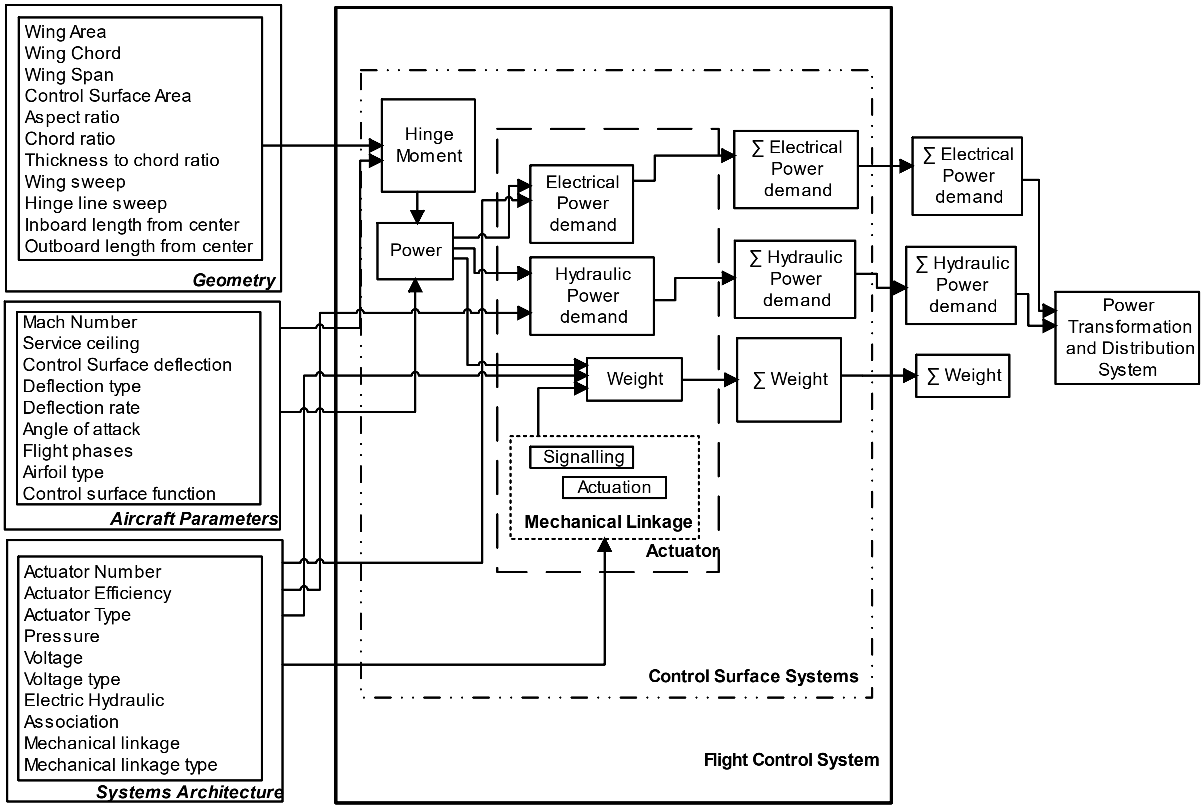


Figure 3-4 Detailed FCS sizing flowchart

The DATCOM method [56] is popular for estimating hinge moments for traditional symmetric wing layouts. The method relies on empirical data and a series of correction factors to estimate the hinge moment. The method is based on several assumptions, including the wing's aspect ratio, sweep angle, and root chord. The DATCOM method estimates the hinge moment coefficient, C_h based on derivates of coefficient due to angle of attack C_{h_α} and the derivative due to deflection C_{h_δ}. The C_{h_α}, includes empirical correlations to account for control surface thickness ratio τ. The method accounts for Mach number correction using the Prandtl-Glauert rule. While the DATCOM method is reliable for estimating the hinge moment, it may not be suitable for unconventional wing designs. However, the method relies on correlation parameters limited to symmetrical types of airfoils or requires experimental values to replace the correction coefficients. Moreover, the method is highly complicated in terms of parameter adaptability.

The Anderson method [57] consists of a simplified correlation to estimate hinge moment chord correction factors due to the angle of attack. K_α and due to deflection K_δ. These are estimated with known wing-to-control surface chord ratio, c_s/c_w. The correction parameters are used along with the hinge moment coefficient, $C_{h_0}$and coefficient due to deflection ΔC_{h_δ} to estimate the hinge moment coefficient C_h. These parameters are estimated with the known maximum Mach number of the aircraft. As a downside, the method does not consider the real-time deflection angle of the control surface, which is assumed to be an empirical correlation. The NASA method is more flexible than the DATCOM method and can be adapted to unconventional airfoils. However, the method may not be as accurate as the DATCOM method for traditional symmetric wing layouts. The DATCOM method is used to estimate the hinge moment when the wing is a traditional symmetric layout. The traditional NACA symmetric wing geometry is analyzed with the DATCOM method. The NASA method is used when the airfoil is non-symmetric or supercritical.

Table 3-2 shows the hinge moment estimation validation results for a typical medium-sized aircraft. The actual hinge moment is estimated based on the available stall load, stroke, and standard deflection rates available from SAE ARP4253B [58] for the A320 control surface. The results are compared between primary and secondary flight control surfaces. DATCOM method is selected to analyse the aircraft control surface. The error variation of -15% to 37% is noted during the comparison. It should be noted that the error variations are valid for traditional wing configurations, and unconventional wing configurations require more sophisticated physics-based hinge moment estimation to address the variations.

Table 3-2 Hinge moment estimation comparison (A320 aircraft)

Aircraft	Estimated Hinge moment (N-m)	Actual Hinge moment (N-m)	Relative Error (%)
Aileron	3120	2295 [58]	36.9
Rudder	6140	7258 [58]	-15.4
Elevator	4043	3447 [58]	17.2
Spoiler	5979	6366 [58]	-6

The overall power requirement of the FCS is generated by using an individual connected actuator subsystem to the control surface. The Hinge moment estimation (HM_{cs}) and the control surface deflection rate ($\dot{\delta}$) is used to estimate the power demand generated per actuator.

$$P_{max} = HM_{cs} \cdot \dot{\delta} \cdot f_{c,ACT}$$

Typically, only the maximum values HM_{cs} and $\dot{\delta}$ are available in a conceptual design. Therefore, Equation (1) includes a correction factor $f_{c,ACT}$. In a conservative approach, the correction factor can be 1; however, if the conceptual designer has some experience, different factors can be applied to different actuators to reflect the variations better.

The system architecture descriptor is used to determine the type of power demand generation and the technology weight of the subsystem. The systems architecture is crucial in checking the FCS reliability and safety level analysis. The workflow accounts for a simplified assumption where each actuator is assumed to be designed to carry the entire load of the control surface. However, further safety level integration is required to be carried out for a full-scale safety analysis of the architecture. The system architecture and power demand connectivity are shown in Figure 3-5 where existing denotes technologies modelled within the workflow and possible denotes possibility of adding novel technologies. The power type connections are determined using the subsystem power demand request which is to be fulfilled by power transformation and distribution system or energy storage. The current workflow accounts for several actuation technologies such as mechanical actuation, hydro-mechanical actuation (HMA), electro-hydraulic servo-actuation (EHSA), more electric actuators such as electro-hydrostatic actuator (EHA) and electromechanical actuator (EMA). The smaller aircraft use mechanical linkages as the forces on the control surfaces are typically lower and are simpler compared to hydraulic types. The flaps and slats consider Power Drive Unit (PDU), which can be hydraulically or electrically driven. The FCS framework allows additional novel subsystem architecture integration to be easily carried out and analyzed along with the conventional actuation types.

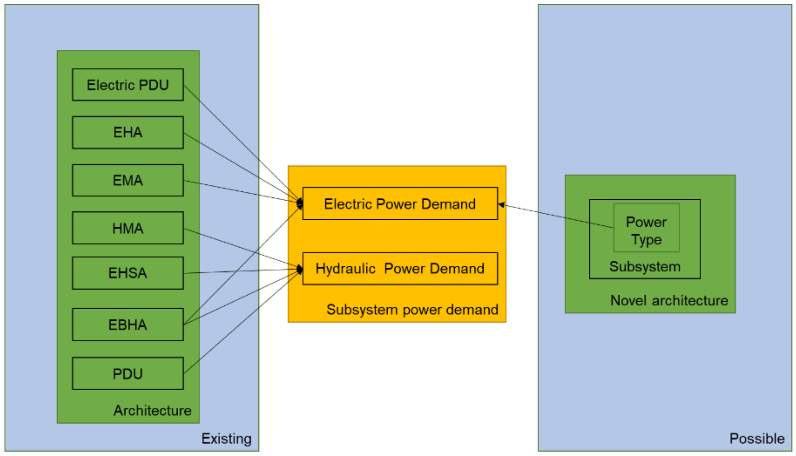


Figure 3-5 FCS subsystems architecture

As discussed earlier, the maximum power requirement is estimated from the hinge moment and the deflection rate. The deflection rate depends on the certification requirements of the control surface. SAE AIR4253B [58] highlights some of the actuator rates for small, medium, and large aircraft. The workflow categorizes the control surface based on its functions. For example, the control surfaces, such as the aileron and spoiler, must be assigned with the roll function. The control surface, which has functionality such as a spoiler used as airbrakes and roll, is to be applied with the function more critical for the aircraft functioning to be assigned. Detailed assignment and the deflection rate highlighted in Table 5-2 in Appendix B.

The overall flight control system is electric, and hydraulic power demand is estimated based on the maximum power demand incurred depending on the flight phase. Although the flight control system usages are highly transient, NASA estimates the hydraulic flow demand for each control surface at four key flight phases: gear retract, flaps retract, stall recovery, and cruise conditions, which are crucial demand phases [57]. The current estimation method categorizes the control surface based on its functions to apply the maximum power demand depending on the flight phase. It is assumed that the key sizing point condition occurs somewhere in one of the flight phases depending on net demand by all the subsystems along with the flight control system. Table 5-3 in Appendix B highlights the hydraulic flow demand and electric power demand applied to the control surface depending on the function and flight phase.

A mechanical push-pull rod actuation is a type of flight control system that uses mechanical linkage to transmit pilot input to the control surfaces. The pilot's input is transmitted through the cockpit controls connected to the push-pull rods. These rods, in turn, are connected to the control surfaces, such as the ailerons, elevators, and rudder. When the pilot moves the controls, the push-pull rods transmit the movement to the control surfaces, causing the aircraft to change its position or direction. Even with a more electric and hybrid-electric architecture, smaller aircraft can potentially still completely contain mechanical surface actuation (due to low hinge moments for low-speed operation or small control surfaces). Therefore, the presented framework includes the weight estimation of a mechanical push–pull rod actuation. This is adapted from the method presented by Torenbeek [26]. The Torenbeek method is based on the aircraft's Maximum Take-off Weight (MTOW) and provides preliminary weight estimates of the FCS. However, the current workflow reformulated the method based on the control surface area (S_{cs}) as a major input to estimate the weight of the actuation. The mechanical backup linkages are considered to be signal-based actuation, which are considered for hydro-mechanical actuators. The signalling requires less load on the linkages. Hence, the overall weight of the FCS is considerably reduced. However, this accounts for servo actuation, which increases the weight of FCS. The mechanical backup linkages can also be assigned to the hydraulic actuation with mechanical signalling based on information presented in SAE ARP5770 [59]. Based on this standard, the primary flight control mechanical linkage cannot be less than 1/8 in; this allows for a reduction in linkage weight of about 12% when the application is limited to signaling. This is further applied to the current estimation method if the mechanical linkage is assigned for signalling. The mechanical actuation weight estimation methods are presented in Table 3-3.

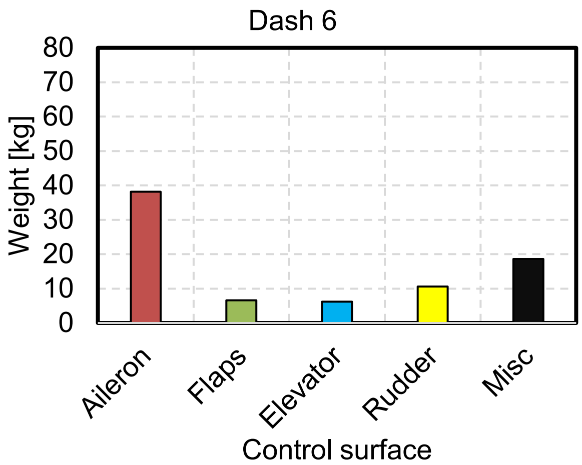
Table 3-3 Weight estimation for mechanical flight control system

Control Surface Type	Control Surface Function	$W_{mech,link}\ (kg)$
Aileron, Spoiler	Roll	$0.0256 \cdot\ (14445 \cdot S_{cs} - 247)^{0.67}$
Rudder	Yaw	$0.0256 \cdot (13814 \cdot S_{cs} - 12708)^{0.67}$
Elevator	Pitch	$0.0256 \cdot (9112.9 \cdot S_{cs} - 12098)^{0.67}$

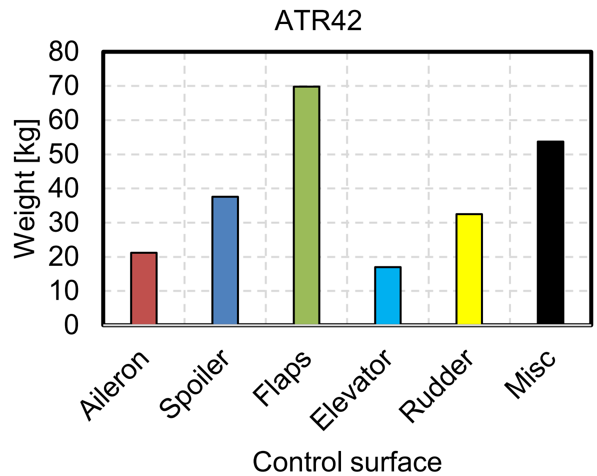
The standard weight estimation of electric actuators is considered using the Power-to-Weight ratio (P/W) method. The electric actuator requires localized electronics to control the power and actuation signalling. Appendix Table 5-4 highlights some considerations to estimate the FCS weight. Additionally, 30% of weight due to miscellaneous equipment is assumed for the entire FCS. The input outputs of the FCS sizing can be found in Table 3-5.

Figure 3-6 shows the FCS subsystem breakdown per control surface for small and medium aircraft. The Dash 6 and ATR42 consist of fully mechanical linkages for the primary flight control surface. Additionally, ATR42 consists of hydraulic spoiler actuation, which is assigned as a roll function. Compared to ATR42, Dash6 has a higher weight for aileron actuation. This is due to the reduced aileron area for ATR42. Additionally, ATR42 consists of hydraulic spoiler actuation, which adds additional weight. Other primary flight controls, such as the elevator and rudder of Dash6 and ATR42, share similar weight percentages, such as 8% and 13% to 14%, respectively. The ATR42 consists of hydraulically powered flaps, which add up weight compared to electrical flaps for Dash6. A combination of aileron and spoilers for medium-sized aircraft share about 14 to 16% weight ratio. It should be noted that B737 consists of added mechanical linkage for the aileron and elevator. Hence, the weight percentage for the control surface shares is higher. However, both A320 and B737 consist of added mechanical linkages for rudder actuation. Hence, both share 17% for ruder actuation. A320 consists of hydraulic-driven horizontal stabilizer actuation, which takes up considerable weight compared to the electrical actuation type. The weight estimate of A320 THSA is about 69.8 kg, while the published weight estimate is about 55 kg [60], which refers to about 26% error comparatively. It is to be noted that other aircraft electrical actuation types are applied with electromechanical actuation, and the power-to-weight ratio is applied with the existing EMA P/W ratio as per data availability.

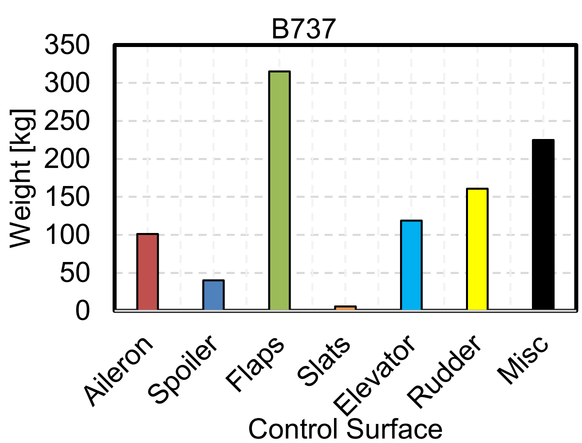
Table 3-4 compares the weight estimates of the flight control system for small and medium-sized aircraft. Smaller aircraft, such as Dash6 and ATR42, are estimated at about 23% error, while medium-sized aircraft incur about 10% error for the FCS weight estimates. Multiple factors involved in the errors go back to the errors within the model and the errors from the inputs. The model errors are the simplified estimation of the hinge moment and the parameter values for the P/W ratio available publicly, which is considered an internal parameter. The input parameters, such as control surface area and location of the control surface, incur small errors, adding up to the estimation errors. Table 3-5 shows the input and output parameters required to size the flight control system.



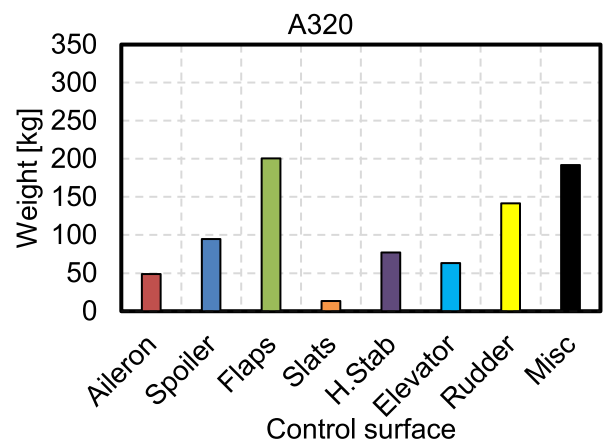
(a) Dash 6 aircraft



(b) ATR 42 aircraft



(c) Boeing 737 aircraft



(d) Airbus A320 aircraft

Figure 3-6 FCS subsystem breakdown of small and medium aircraft

Table 3-4 Flight control system weight comparison for smaller and medium sized aircraft

Aircraft	Weight Estimated (kg)	Weight Actual (kg)	Relative Error (%)
Dash6	80	65 [61]	23.4
ATR42	214	195 [61]	20.8
B737	967	1065 [31]	-9.1
A320	931	772 [61]	7.7

Table 3-5 Flight control system input/output parameters

Attribute	Parameter
Internal Parameters	
Subsystems	Efficiency P/W ratio
External Parameters	
Geometry	Wing area Wing chord Wingspan Aspect ratio Control surface area Chord ratio Thickness to chord ratio Service Wing sweep quarter chord Hinge line sweep Control surface location
Systems Architecture	Control surface definition Control surface function Pressure Voltage Voltage type Actuator type
Aircraft	Mach Service ceiling Maximum Deflection Angle of attack Control surface deflection rate

3.3.2 Hydraulic System

In a conventional aircraft, the FCS and landing gear are the major consumers powered by the centralized hydraulic system. Typically, there are one to two independent hydraulic systems in an aircraft. The hydraulic distribution system fulfils the hydraulic flow demand by the consumers. The hydraulic distribution system consists of hydraulic tubing and specific hydraulic fluid, typically the same for all consumers. The hydraulic distribution sizing is based on the maximum hydraulic flow demand considered across the multiple flight phases. The hydraulic fluid requirements are fulfilled by the reservoirs which store the hydraulic fluid. The hydraulic fluid sizing depends on the globally set hydraulic fluid type and the internal power density of the hydraulic fluid. The hydraulic generation includes Engine Driven Pumps (EDP), which transforms engine shaft power into hydraulic power, and Electric Motor Pump (EMP), which transform the electrical power to run the hydraulic pump. The sizing considerations for the EDP and EMP are based on the available systems to fulfil the hydraulic demand and the power split between the pumps. This section will explore the detailed sizing methods for all the hydraulic subsystems.

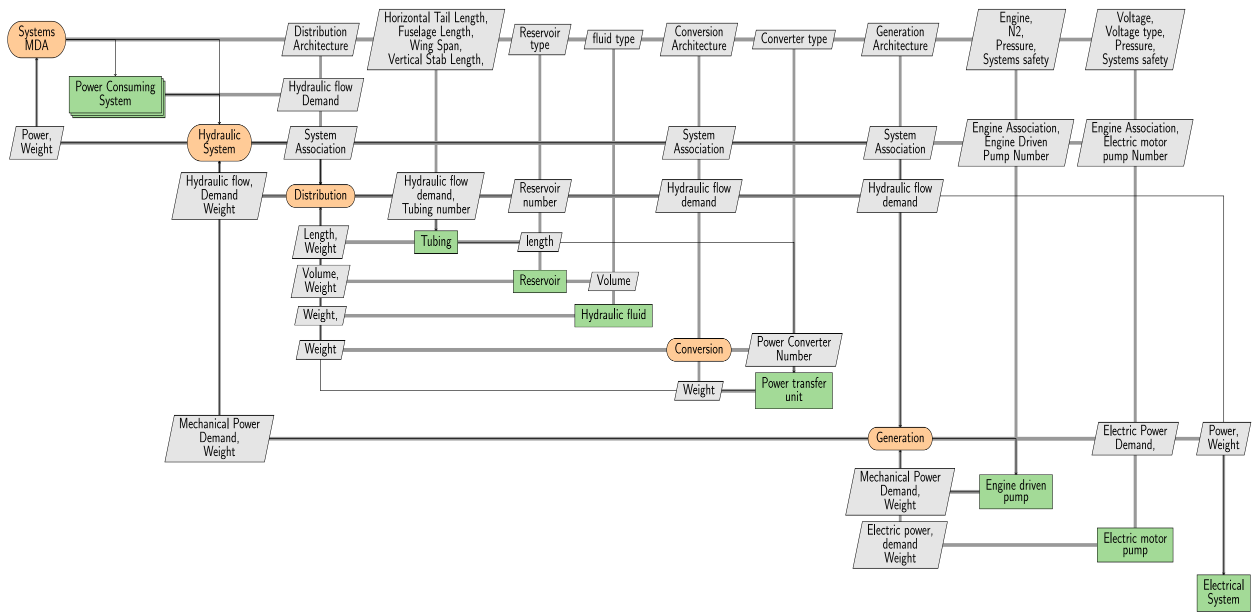


Figure 3-7 Hydraulic System sizing workflow overview

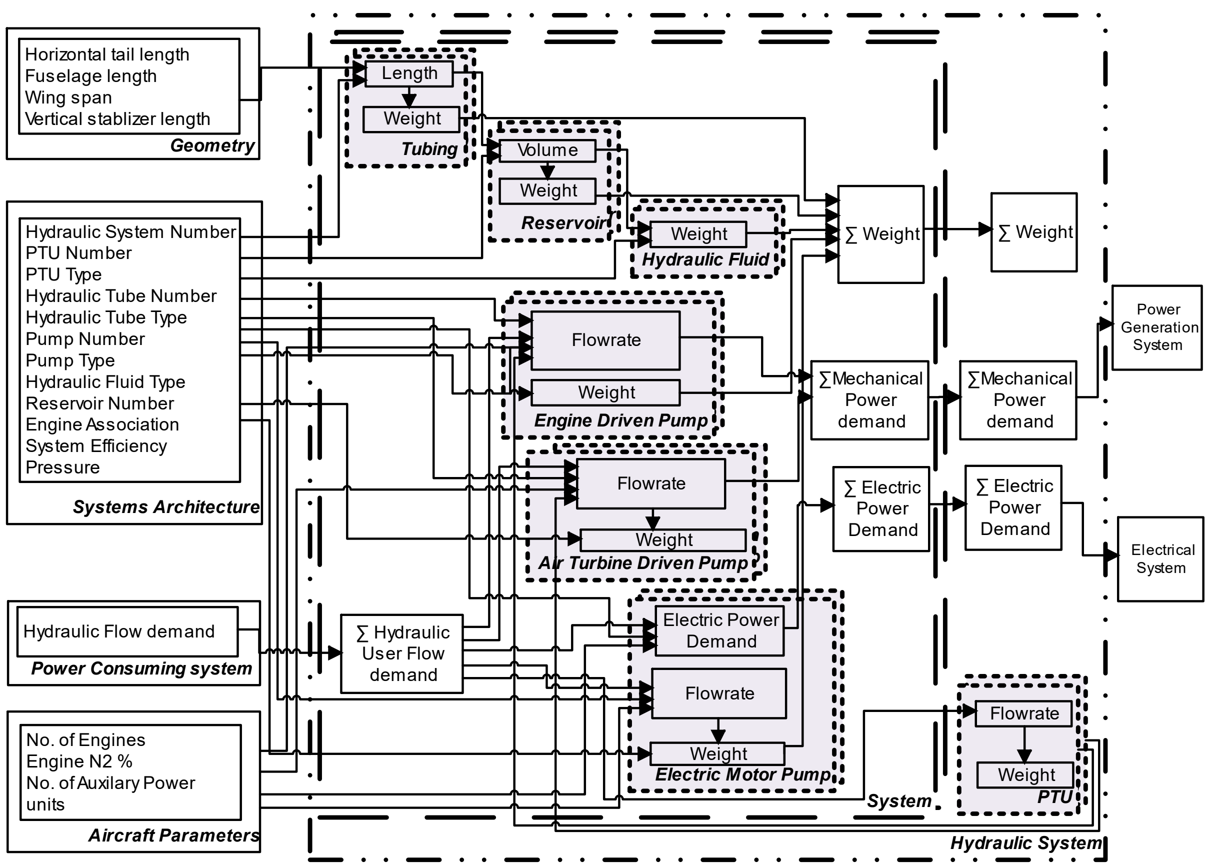


Figure 3-8 Detailed hydraulic system sizing flowchart

One of the major assumptions considered within the hydraulic system is the power split between EDP and an EMP to estimate the available systems to fulfil the consumer flow demand. The total flow demand of the individual hydraulic systems, $\dot{Q}_{Hyd,sys,i}$ is supplied by the associated hydraulic pumps, typically one EDP and one EMP, in conventional aircraft. For more than one EDP,

$$\dot{Q}_{EDP}=\dot{Q}_{Hyd,sys,i}/N_{EDP} \tag{3-1}$$

where N_{EDP}, is the number of EDPs available within the hydraulic system. The split factor, f_{split}, can be assumed as a factor in split flow to the EMP. As per SAE ARP6277 [62], if the EMP is set as backup, it only compensates the flow when the engines are at idle which corresponds to

$$\dot{Q}_{EMP}= f_{split}\ \dot{Q}_{Hyd,sys,i} \tag{3-2}$$

where f_{split}= 0.16 for a traditional hydraulic system architecture where the EMP is set as 'backup' source. From Figure 3-9, it is evident that the total flow of EMP and EDP during ground idle situation is equivalent to flight idle scenarios.

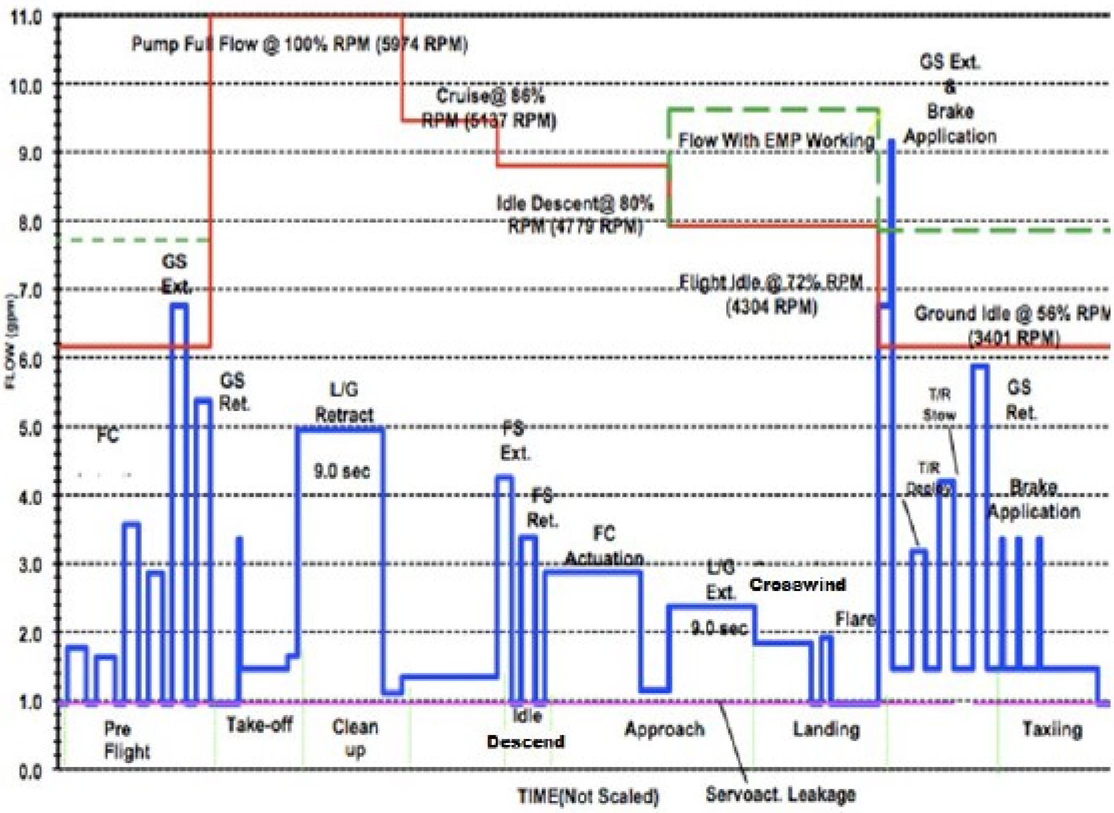


Figure 3-9 Hydraulic system flow demand [62]

In a more electric architecture, removing EDPs leads to an EMP set as the main instead of backup. It is also possible that EMP is set as main instead of backup in a particular architecture. In these scenarios, $\dot{Q}_{EMP}$ is equal to $\dot{Q}_{Hyd,sys,i}$, which corresponds to $f_{split} = 1$. It should be noted that in the case where EDP and EMP are present, setting the $f_{split} = 1$, could lead to an oversizing scenario. Hence, the value can be set by applying several safety scenarios specific to the hydraulic system. For example, if there are two EDP and two EMP, all set as main source, design expert can explore possibility of f_{split} between 0.16 and 1 to meet safety targets and would give less system weight.

For a conventional aircraft, the electric motor component of EMP is generally driven by the electrical generators which are connected to the engine. Hence, the power demand is fulfilled by the electrical system. The electrical system fulfils the EMP request for Electric Power Demand. The flow demand is converted as a power request with a pressure rating to build in the hydraulic system. Normally, the global pressure is used for the conversion.

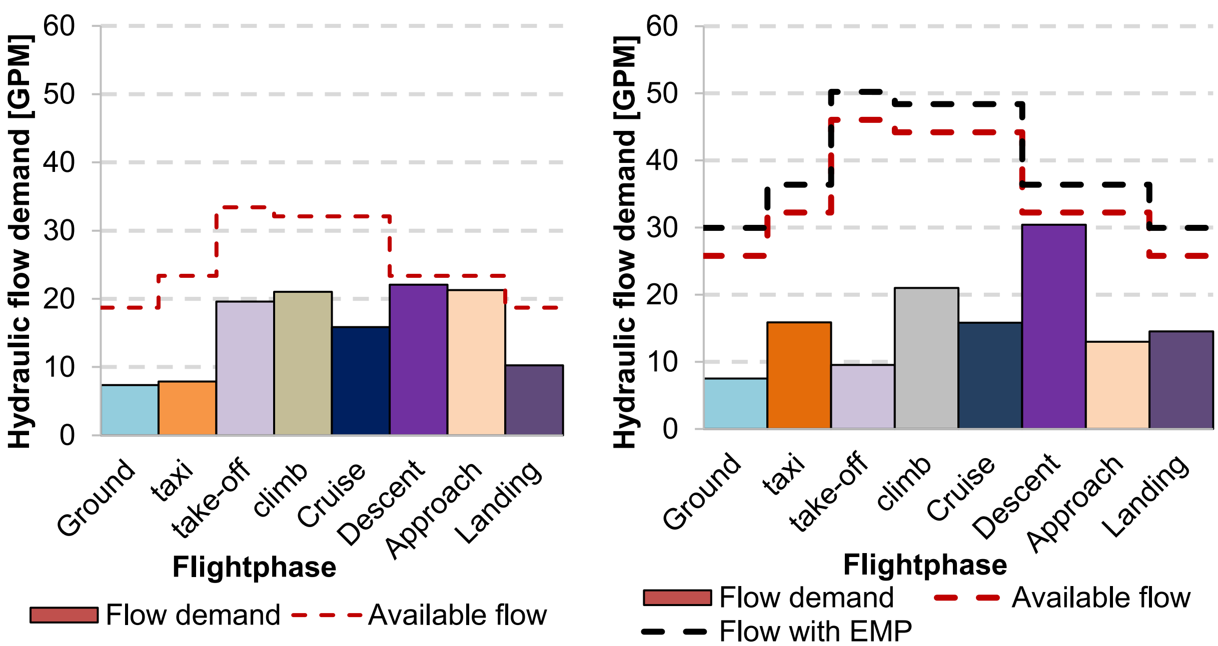
$$P_{EMP\,flightphase} = \dot{Q}_{EMP\,flightphase}.\Delta p_{sys} \quad (3\text{-}3)$$

For example, for sizing the EDPs, equation (3-4) can be used, where, $\dot{Q}_{Eff}$ is the adjusted available flowrate considering the engine operation with respect to the flight phase and $n_{ENG,shaft,\%}$ is the associated shaft speed percentage of the engine, and the EDP efficiency η_{EDP}:

$$\dot{Q}_{Eff_{EDP}}(flightphase) = \frac{\dot{Q}_{EDP}(flightphase)}{n_{ENG,shaft,\%}(flightphase).\eta_{EDP}} \quad (3\text{-}4)$$

Once the EDP is sized, the engine shaft power off-take can be estimated.

Figure 3-10 shows the hydraulic flow demand for a typical medium-sized aircraft. As the hydraulic engine-driven pumps are connected to the engine shaft, these are subjected to the operational envelope of the engine. Hence, the available flow corresponds to the maximum flow available at a particular flight phase. The flow demand corresponds to the maximum hydraulic flow demand request by all the connected hydraulic subsystems to the hydraulic system. It is to be noted that the descent phase has the highest flow demand for A320 hydraulic system A and system B with minimum available flow from the pump. This corresponds to the sizing point for the hydraulic system. System B consists of an additional backup electric motor-driven pump, considered a backup to add additional flow to the hydraulic system. Hence, the total flow available corresponds to the flow with the electric motor pump.



(a) System A hydraulic flow demand (b) System B hydraulic flow demand

Figure 3-10 A320 hydraulic flow demand

Table 3-6 Hydraulic subsystems flow demand comparison

Aircraft	Hydraulic System	Subsystem	Estimated flow rating (GPM)	Actual flow rating (GPM)	Relative Error (%)
A320	System A	Engine Driven Pump	33.4	37	-9.7
A320	System B		46	37	24.5
B737	System A		25.7	22.5	14.1
A320	System B	Electric Motor Pump	4.2	6	-30.5
B737	System B		12	5.7	110.5
B737	System C		2	3.1	-34.2

Table 3-6 shows typical aircraft hydraulic subsystem sizing variations. A320 and B737 hydraulic generation subsystems, such as EDP and EMP sizes for different hydraulic systems, are compared to the publicly available actual rated flow. The Boeing 737 has two EDPs in System A. Hence, the total flow is shared between the two EDPs. However, the A320 has one EDP per hydraulic system, and the flow demand of System B is higher than that of System A. Overall, minimum and maximum error corresponds to -10% and 25%, respectively, for EDPs. In the case of EMPs, the sizing depends on the subsystem assignment. It is assumed that the standalone system is considered a main system. This corresponds to additional errors for the subsystem. Additionally, the magnitude of the flow of backup EMP is very low compared to the EDP, which is highly evident from the error variation between -35% and 110%. It is to be noted that although at the subsystem level, the error variations are high, at the system level, this adds up to a minimum error increment.

For EMPs, the required flow demand must be translated into electrical power demand, which impacts the sizing of the electrical power systems, discussed in the following subsection. All efficiency and power-to-weight ratio assumptions for the weight estimation of the pumps are provided in Appendix A.

The EDPs are connected to the engine, providing power to drive the subsystem. These are typically through the drive shafts, hence, mechanically powered. The net shaft power requirement generated by the EDP is used to size the Pump and to carry out mission-level analysis and can be calculated as follows:

$$P_{shaft,hyd} = \dot{Q}_{Eff_{EDP,i}} . \frac{\Delta p_{sys}}{\eta_{gb}} \tag{3-5}$$

Where, P_{shaft} is the engine shaft power requirement and η_{gb} is the gearbox efficiency. For simplicity, sizing assumes Δp_{sys} to be the maximum pressure rating of the system. Table 3-7 shows the overall input and output parameters for the hydraulic system estimation.

Table 3-7 Hydraulic system input/output parameters

Attribute	Parameter
Internal Parameters	
Power Consuming System	Hydraulic power demand Hydraulic system association
Subsystems	Efficiency P/W ratio
External Parameters	
Geometry	Fuselage length Wingspan Vertical stabilizer length Horizontal stabilizer length Location consumers
Systems Architecture	Tubing number Hydraulic fluid type Number of EDP Number of ATDP Number of EMP Number of RAT System pressure Engine association Number of reservoirs Voltage Voltage type
Aircraft	Engine n2% Number of engines Number of aux power unit

3.3.3 Electrical System

The electrical systems consist of subsystems responsible for generating, converting and distributing electrical power supply to all the subsystems within the aircraft. The electrical distribution includes feeders responsible for carrying electrical power to the consuming systems. The simplified estimation allows the categorization of the feeders based on voltage, voltage type and the classification of the feeders. The major classification is based on whether the feeders provide electrical power to the consumers or if they carry the electrical power from the generation. Instance generation is solely based on these criteria. The electrical power generation systems consist of Engine-driven generators, which convert the mechanical power from the engine to electrical power. The instance generation of the generators is solely based on the architecture, including the voltage and voltage type of the generators. The power converters are an interface between the power generation and distribution system. The power converters categorization is based on the type of power conversion. However, it is assumed that each feeder classification needs to be interfaced with the power converters. The section will explore detailed sizing methods for all the subsystems.

The current electrical system framework falls within the power transformation and distribution system, as shown in Figure 3-11. The overall structure of the electrical system sizing methodology is based on the approach developed by Liscouët-Hanke. However, more detail is added to the distribution and conversion system, which consists of feeders and power converters. Alternating current (AC) feeders and direct current (DC) feeders are modelled with multiple voltage assignments to estimate the entire load on the feeder based on consumer assignments. The minimum requirement for analyzing electrical system modules is the availability of details of the power-consuming systems. The power-consuming systems accumulate the subsystems which require power from the power transformation and distribution systems. The electrical system is subdivided into Distribution and Conversion as well as Generation systems specifically for electrical generation. The Distribution and Conversion systems consist of feeders and power converters. The power demands from the power-consuming system are categorized based on the architecture, voltage and voltage type of specific distribution and conversion systems. In a typical aircraft, left and right distribution systems provide electrical power to specific consumers. The power-consuming system subsystems corresponding to the connected distribution system are used to estimate the net electric power demand for the distribution and conversion system. The specific feeders are sized as per the net user power demand within the distribution conversion system.

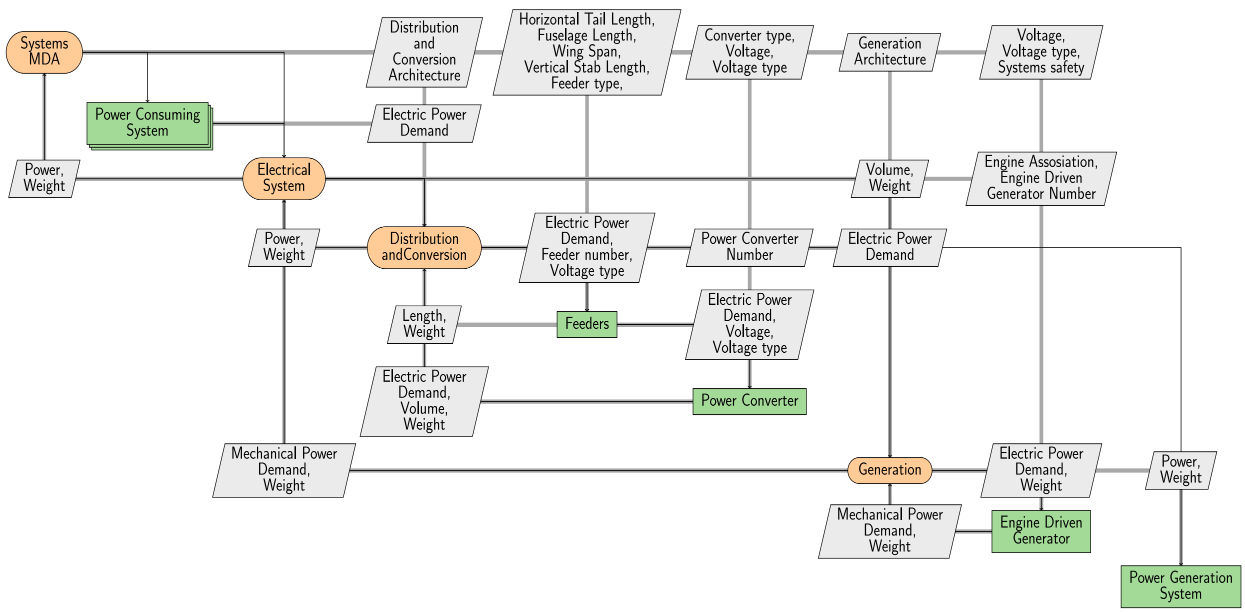


Figure 3-11 Electrical systems sizing workflow overview.

The feeders are responsible for distributing the electrical power supply to the various systems and subsystems within the aircraft. The feeder module consists of AC feeders and DC feeders, with multiple voltage assignments to estimate the entire load on the feeder based on consumer assignments. The SAE AIR6540 [63] document is used to estimate the feeder weight, considering voltage and voltage type variations. Additionally, the SAE AS50881 [[64] document is used to model different sizes of feeders.

The aircraft feeders are exposed to several physical and environmental factors, including high altitude and low temperature, which may affect their properties, such as mechanical strength and conduction properties. This will primarily be evident in terms of voltage drop across the conductors; hence, the selection of wires must consider these parameters to increase efficiency by reducing the voltage drop across the feeders. The selection needs to consider the maximum current carrying capacity of the conductors to minimize the voltage drop across the feeders.

The initial step is to estimate the overall length of the wiring requirement. The workflow accounts for the feeders for power consumption and generation systems. These feeders provide power from the power generation system (such as engines or APUs) to the power transformation and distribution system (i.e., electric generators or EMPs). The basic consideration given for the length estimation is the actual location of the generator within the engine from where the wiring starts.

$$L_{Wire,eng} = z_{gen} + x_{gen} \tag{3-6}$$

Where, $L_{Wire,eng}$ is the wiring length from the engine to the engine attachment with the wing or fuselage; z_{gen} and x_{gen} are the coordinates of the generator. If the location of the generator is unknown, $L_{Wire,eng}$ can be assumed as the sum of half the diameter and half the length of the engine, which applies to any type of engine attachment, including blended wing body aircraft configuration with embedded engines.

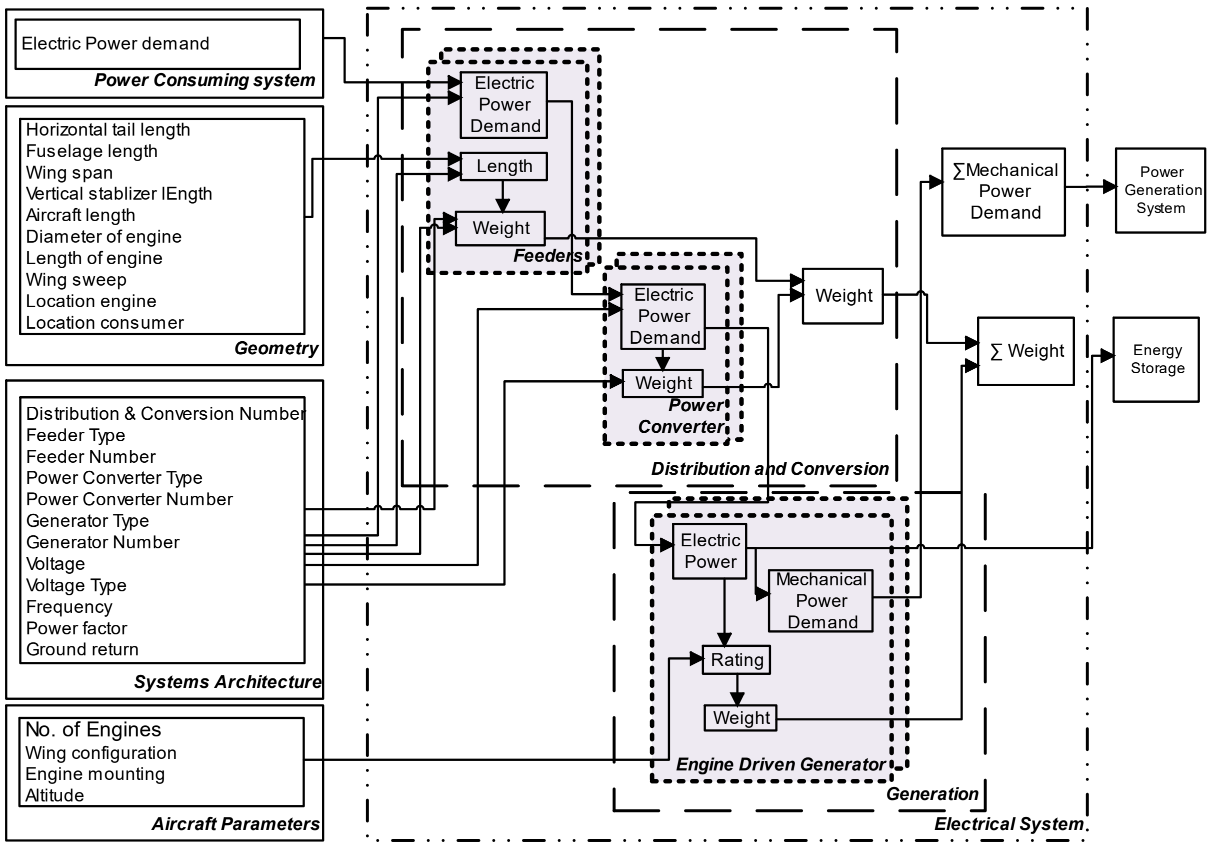


Figure 3-12 Electrical system sizing detailed flowchart

To represent the wiring length from the generation to the distribution conversion $L_{wire,gen-elec}$, the subsystem location factor $f_{location}$, is introduced as the ratio of the distance between generation and conversion to the fuselage length. The wiring length can be calculated according to Eq.(3-7) as a function of the fuselage length, $L_{fuselage}$:

To represent the wiring length from the generation to the distribution conversion $L_{wire,gen-elec}$, the subsystem location factor $f_{location}$, is introduced, which is the ratio of the distance between generation and conversion to the fuselage length. The wiring length can be calculated according to Eq.(3-7) as a function of the fuselage length, $L_{fuselage}$:

$$L_{wire,gen-elec,fuselage} = f_{location} \cdot L_{fuselage} \tag{3-7}$$

For subsystems such as APU, which is traditionally placed at the aft fuselage, $f_{location} = 0.9$ can be used; for fuselage - mounted engine $f_{location} = 0.8$ can be used for simplicity. If the battery is placed near the power conversion system, a value of $f_{location} = 0$ is to be considered to eliminate the length. The maximum value of, $f_{location} = 1$ is applicable to a tail-mounted engine configuration with an electrical subsystem placed in the front fuselage. For a wing-mounted engine configuration, a value of $f_{location} = 0.5$, can be used for simplicity. Additionally, the subsystems placed within the wing need to consider the wiring length in the wing.

For low-wing configurations, the length of wiring is:

$$L_{wire,eng-elec,wing} = \frac{y_{eng}}{\cos(\Lambda)} \tag{3-8}$$

Where y_{eng} is the distance of the engine from the aircraft center axis along the y-axis, and Λ is the wing sweep angle,

For high-wing configurations, one needs to account for additional wiring in the fuselage, considering the fuselage diameter, $D_{fuselage}$:

$$L_{Wir\ eng-elec} = \frac{y_{eng}}{\cos(\Lambda)} + D_{fuselage} \tag{3-9}$$

For the consumer wiring, it is assumed that the feeders power several subsystems along the fuselage, accumulating the power demand, which the power conversion system fulfills. Hence, except for the flight control system, the wiring is assumed to be running across the fuselage. The critical parameters for the flight control system are the electric subsystem in the wings and tail section, which triggers the wiring length extension to the subsystem location. Equation (3-7) can be adapted to estimate the length of consumer wiring. However, the parameter $F_{location}$ will need to consider maximum distance between connected subsystems. For the consumer system within the wing, equations (3-8) and (3-9) can be adapted to replace y_{eng} with $\max(y_{consumer_i})$ connected to feeder within the wing.

The aircraft can use 'AC' and 'DC' types of generation systems. The 'DC' system consists of a two-wire setup, and the three-phase 'AC' uses a three-wire setup. The 'DC' system benefits from using the airframe as a ground return carried by one of the wires. This kind of architecture is modelled by halving the length of the wires. The high voltage return line considerations must meet the certification requirements due to high electromagnetic interferences and safety hazards. Additionally, composite structures pose a high resistance to the return line, requiring detailed study to enable ground return capability. These cases must consider a dedicated return line for the aircraft system wiring.

$$L_{Wiring,DC_{ground\ return}} = \frac{1}{2} L_{Wiring,DC} \tag{3-10}$$

Where, $L_{Wiring_{ground\ return}}$ is the wiring length when the ground return is possible, typically in a 'DC' system measured in m.

The next step of the wiring load estimation is the effective current, I_{eff} , estimation. The parameter is represented as a function of the Bundle derating factor, $f_{\ Bundle\ LF}$ and altitude derating $f_{Altitude\ DF}$. The parameter is, in turn, a function of the number of wires and the bundle loading. About 70% of the derating factor is noticed with a bundle loading between 80% to 100% bundle loading for a two to three-wire setup. For simplicity, a conservative value of 100% loading can be chosen which can be conservative. The altitude derating factor $f_{Altitude\ DF}$, is estimated using the curve fitting method, altitude being the input parameter. The service ceiling of the aircraft is used to estimate the $f_{Altitude\ DF}$.

$$I_{eff} = \frac{I}{(f_{\ Bundle\ LF} \times f_{Altitude\ DF})} \tag{3-11}$$

Where, I_{eff} is the effective current, $f_{\ Bundle\ LF}$ is the bundle derating factor, $f_{Altitude\ DF}$, is the altitude derating factor. The I_{eff} , which includes the current correction that can be used to select the wire size as per SAE50881 wire sizes in terms of AWG.

The next step is to estimate the losses in the wiring in terms of the wire resistance. The wiring resistance W_{res} for a DC wire is estimated simply as a function of the overall length L_{Wiring}, individual feeder and temperature difference is used to estimate the W_{res}. The DC wire has a uniform distribution of current throughout the cross-section. However, in an AC feeder, due to the frequency effect, the current travel in the outer edge has a non-uniform current distribution called the skin effect. To account for the resistance correction, mr is estimated as a function of resistance and frequency, and Rac is estimated as a function of mr. Finally, a simplified estimation of AC inductance X_{ac} , is estimated in terms of frequency and wire size. The SAE method for AC wire is valid until 1 kHz.

For a DC feeder, the wire resistance and the current are used to estimate the Voltage drop, V_{drop}. However, the AC feeder undergoes additional power factor pf , X_{ac} , and Rac correction to estimate the voltage drop. The net effective Voltage will be the V_{drop} added to the V_{Wire}.

$$V_{Wire,eff} = V_{Wire} + V_{drop} \tag{3-12}$$

Where, $V_{Wire,eff}$ is the effective voltage estimated in V, V_{drop} , is the Voltage drop across the wiring length in V.

The power on the feeders can be estimated as follows: -

$$P_{Wiring} = I_{eff}.V_{Wire,eff} \tag{3-13}$$

Power converters are the units used to convert the incoming current into the type and voltage the power-consuming systems request. The conventional aircraft, consisting of a 28VDC system, typically consists of DCDC and ACDC converters providing DC and AC supply. The architecture possibilities of the modern aircraft consist of high voltage DC systems such as 270VDC and high voltage AC systems typically more than 230 VAC. The section will discuss the power converter architecture modelling adapted within the framework.

The power demand of the power converters is considered with the efficiency of individual types of power converters. The weight estimation of the power converters is based on the type of the power converters and the individual P/W ratio.

$$P_{conv,i} = \frac{P_{Wiring,i}}{\eta_{conv,i}} \tag{3-14}$$

The first type of power converter is ACAC type, converting AC current to required AC with variable voltage variations required by the power consumers. In a conventional aircraft, subsystems such as ECS, Galley and Lights typically use AC. As discussed in [65,66], the ACAC type of power converter type consists of a rectifier unit to convert the incoming AC to DC form. The power factor correction (PFC) unit is used as a filter to reduce the harmonics after the conversion. The DC links are then used as a connection point to the inverters, which convert the DC to AC. The ACAC converters P/W are found to be about 12 kg/kW when the power is less than 5 kW and 1.43 kg/kW when the power is more than 5 kW [66].

The ACDC converters typically supply the DC loads consumers. The FCS and Avionics are typical DC consumers within a conventional aircraft. This type of converter consists of a rectifier unit and PFC similar to the ACAC converters. However, the heavy inverter unit is replaced with the DCDC converters. Overall, the elimination of the inverters improves the P/W ratio of the ACDC converters compared to the ACAC type of converters. ACAC converter P/W ratio is found to be about 7 kg/kW when the power demand is less than 5 kW and about 0.51 kg/kW when the power is more than 5 kW [66].

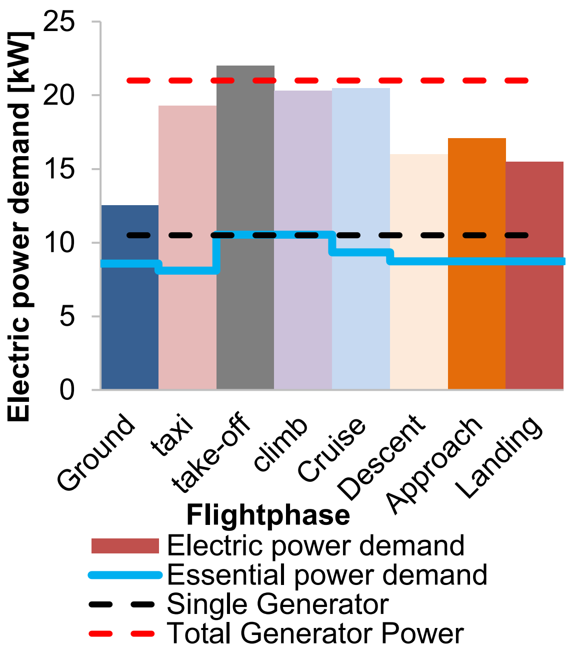
The DC/AC converters eliminate the rectifier and associated PFC units and DC links, which considerably improves the P/W ratio of this type of power converter. The DCAC units consist of inverters to convert DC to AC, and PFC is used as a filter. However, the PFC does not require harmonics correction and has improved power density. Overall, the DCAC type of converters have a P/W ratio of 6 kg/kW when the power demand is less than 5 kW and 0.95 kg/kW when the power is higher than 5 kW [66]

The DC/DC converters consist of only a couple of power electronics which have high power density. Hence, these are considered to have the highest P/W ratio compared to all other power converters. The P/W ratio of the DCDC converters is about 1 kg/ kW for a power demand less than 5 kW and 0.17 kg/kW for a power demand greater than 5 kW [66].

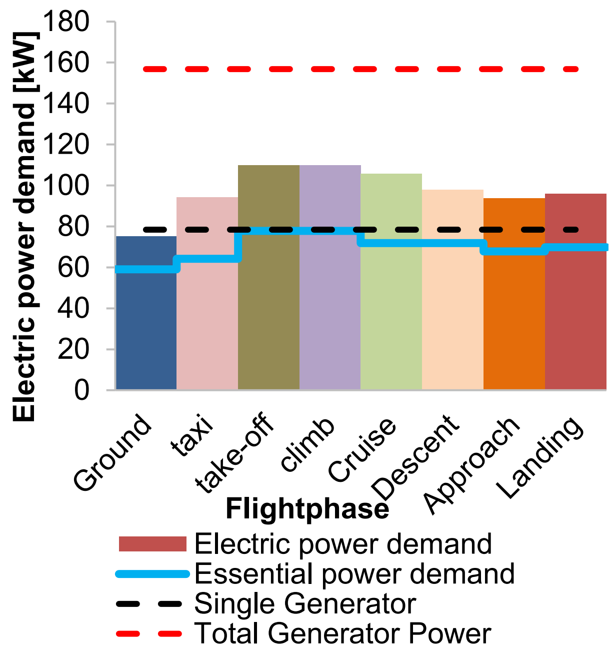
The distribution and conversion accumulate the feeders and power converters within the assigned architecture. The assignments are either based on the (left and right) system or this could be solely based on the feeders and power converters available within the electrical system architecture. However, unlike the hydraulic system, the electrical distribution and conversion system are not independent of the generation system, as per the assumption that any generation system can supply each of the distribution systems. Hence, the sizing of individual generations depends on all the distribution and conversion available. However, during emergency conditions, the sizing depends on the essential loads required to be supplied by the individual generation.

The major input to the generation system is the distribution and conversion system. All the feeder types are assumed to be interfaced with a power converter. Hence, the generation system is interfaced with the power converter. The generation system consists of an AC or DC type of generator. The assignment is as per the electrical generation architecture. One of the major assumptions considered for the sizing of the electrical generation is that each engine-driven generator can supply all the essential loads within the aircraft regardless of the distribution architecture assignments. The minimal safety considerations are given in [7], considering three scenarios: normal operation, single-engine failure, and a combined failure (single-engine off and opposite generator off condition). The engine failure will mean that all the generators connected to the engine will not be operational. For example, if an aircraft with two engines has two generators per engine, engine failure would mean two generators non-operational. Additionally, the third scenario considers additional opposite generator failure to size the generators. In this case, the single generator will have to provide the entire essential load of the aircraft. The finalized size of the generators is weighed using the P/W ratio of the generator as per architecture. Table 3-9 shows detailed input-output for the electrical system sizing workflow.

Figure 3-13 shows the electric power demand request to the electrical system and the available electrical generation to fulfil the power requirements for smaller and medium aircraft. The electrical generators meet the total electrical power demand by the consumers while the interconnection between the distribution system requires either generation to fulfil the energy requirements. Hence, the sizing is driven by the essential power demand requirement of the consuming system. It can be seen that the Dash 6 power consumers request in total about 10.5 kW power as an essential demand while the A320 requests about 78 kW power during take-off and climb phase. The sizing point is chosen during the phase as a major requirement criterion. It is assumed that the electrical power demand for subsystems such as galleys and lights may be shed to avoid generator overloading during the take-off phase of Dash6, where the maximum power demand is slightly higher than the generator rating.



(a) Dash 6 electric power demand



(b) A320 electric power demand

Figure 3-13 Electrical power demand

Table 3-8 compares the electrical generation for smaller and medium-sized aircraft. The aircraft consists of two generators to fulfil the electrical power requirements of the consumers. The smaller aircraft such as Dash6 and ATR42 are considered with the DC low voltage generators, and aircraft such as A320 and B737 consist of AC generators. For comparison, the AC generator's power rating is considered in kW with a power factor of 0.95. The error variation for smaller aircraft ranges to a maximum of 50%. This is due to the uncertainty regarding the consumer power consumption assumptions considered within the workflow. A maximum error of 26% is seen for the A320 case. Comparing similar aircraft such as B737 and A320, A320 generators are larger than the B737. The overall error propagation within the modelling corresponds to the electrical power demand assumption for the consumers and the multiple physical layers of propagation. The physical layers are the feeders and power converters within the electrical distribution and conversion system. Detailed analyses are required to estimate the error propagation within the layers to address generation errors.

Table 3-8 Electrical generation subsystems comparison

Aircraft	Estimated Generator rating [kW]	Actual Generator rating (PF=0.95) [kW]	Relative Error (%)
Dash6	10.5	7	50
ATR42	13.6	12	13.3
B737	54.2	42.8	-8.3
A320	78.4	85.5	26.7

The net power demand by the electrical generators is translated into the engine's secondary power shaft in terms of mechanical power demand.

$$P_{shaft,elec} = \frac{\sum P_{gen,i}}{\eta_{gb}} \tag{3-15}$$

Where, $P_{shaft,elec}$ is the shaft power demand by electrical generators, η_{gb} , is the secondary power gearbox efficiency, and $P_{gen},$ is the power demand of the electrical generator.

The entire electrical system weight is cumulative of the distribution and conversion system and the generation system. Additionally, to account for components such as switches and connectors, 30% of the weight is added to the total weight of the electrical system.

Table 3-9 Electrical system input/output parameters

Attribute	Parameter
Internal Parameters	
Power Consuming System Hydraulic System	Electric power demand
Subsystems	Efficiency P/W ratio
External Parameters	
Geometry	Horizontal tail length Fuselage length Wingspan Vertical stabilizer length Aircraft length Diameter of Engine Length of engine Wing sweep Location engine Location consumers
Systems Architecture	Distribution & Conversion architecture Feeder type Feeder number Power converter type Power converter number Generator type Generator number

	Voltage Voltage type Frequency Power factor Ground return
Aircraft	Number of engines Wing configuration Engine mounting Altitude

3.3.4 Other Systems

The other power-consuming systems can be subdivided based on the characteristics of power consumption type. The power consuming systems which consume electrical power includes galleys, entertainment, furnishing, lights, avionics, instruments, air conditioning fans, and fuel pumps. Conventional aircraft consisting of air conditioning and ice protection systems that require bleed air flow are categorized as pneumatic power consumers. Moreover, the landing gears are hydraulic users. However, the subsystems are electrified in a more electric and all-electric configuration and request electric power demand. In these cases, these are categorized as electric power consumers.

The conventional hydraulic system is designed to a specific global pressure level. For example, the subsystems are either 3000 psi or 5000 psi as per the aircraft hydraulic system architecture. However, in the case of electrical system users, the consumers are designed to have multiple voltages and types. The consumers are to specific AC or DC type and specific voltage level feeders as per the specific subsystem request. As discussed in the previous section, this assignment is further used for power transformation and distribution system analysis. The current workflow voltage assignments are limited to the data availability. The section discusses the detailed sizing of various other power-consuming systems.

Galley Entertainment and Furnishing

The galley, entertainment and furnishing correspond to the most significant weight and electrical power consumer within aircraft systems. The galleys include the subsystems and structures used to prepare, store, and serve food and beverages for the passengers. These include ovens, refrigerators, coffee makers, water boilers, storage compartments, etc. These are significant depending on the number of passengers to serve. The galley sizing can differ for business jet aircraft compared to commercial and smaller aircraft, depending on the customer's requirements.

The furnishing consists of aircraft interior arrangements, including passenger seats, walls, flooring, and storage compartments, except for galleys. These can also include lavatories and other subsystems responsible for the functionality of the cabin. The commercial jet consists of standard furnishing to fit maximum passengers. However, for a business jet, the number of passengers is compromised to fit in a visually appealing environment for the specific number of passengers specified by TLAR. These can include premium seating, floor carpeting, and branding elements to provide a premium experience for the passengers. The furnishing is a significant contributor to the aircraft system weight.

The aircraft entertainment includes subsystems providing entertainment and crew information to the passengers to enhance the flight experience and stay engaged while on board. Typically, a commercial aircraft includes seatback screens or shared overhead screens. Depending on the customer requirements, the business jet version can include larger screens and even projection subsystems. The business jet can also feature advanced Wi-Fi connectivity for customers who want to work onboard. Personal devices and connectivity are yet another feature adding power requirements by the entertainment.

The current workflow considers the power requirements of the galleys, furnishing and entertainment with respect to the maximum power requirement certifiable. It is to be noted that the power requirements of these subsystems are highly variable depending on the customer requirement and require intense studies. In the business jet variant, the level of entertainment requirement is highly customized and depends purely on the customer's needs. However, these loads can be shed during emergency conditions when considered non-essential (this might differ for business aircraft operations). The variability in the essential loads can be prefixed with a ratio of shed load with respect to maximum power demand. The value can range from 0 to 0.5, depending on the aircraft type. As a simplified estimation, the current workflow adapts the methodology established by Esdras and Liscouët-Hanke [67], to estimate the power requirement. The nominal electric power demand for these systems $P_{GEF,nom}$ is defined in Eq. (3-16) as a function of the fuselage length $L_{fusleage}$, fuselage width $W_{fuselage}$, and the number of engines N_{eng}.

$$P_{GEF,nom} = 10.284e^{0.0139\left(\frac{L_{fusleage} \cdot W_{fuselage}}{N_{eng}}\right)} \tag{3-16}$$

The power demand of the subsystems can vary depending on the flight phase. The nominal power demand, $P_{GEF,nom}$, must be multiplied by the usage factors provided in Appendix B. The usage factor translates the nominal power demand of the subsystem to the actual power demand incurred during the flight phase with a ratio between zero to one. Table 5-7 Subsystem power demand ratios, to obtain the power demand per flight phase.

The galley, entertainment, and furnishing contribute significantly towards the aircraft's overall weight. The method used by NASA Flops [30] has applications specifically to larger aircraft and overpredicts the weight of these systems for smaller commuter aircraft. The current workflow considers the weight build-up of the subsystems based on the number of passengers, the number of crew, and cabin volume. Smaller aircraft, such as Dash 6 or Cessna aircraft, typically have front and aft baggage compartments and different cabin layouts are enabled by trading passenger seats with baggage area. Therefore, the method presented here considers the front and aft baggage volume and cabin volume to estimate the net weight of the galley, entertainment, and furnishing subsystems. The weight of the galley, entertainment, and furnishing subsystem W_{GEF} (kg) can be estimated using Equation (14), with N_{crew} as the number of crew members, N_{pax} as the number of passengers and V_{cabin} as the cabin volume in ft^3 and the factor k_{GEF}:

$$W_{GEF} = k_{GEF}.\frac{\left(N_{crew} + N_{pax}\right)^{1.65}}{V_{cabin}{}^{0.18}} \tag{3-17}$$

where k_{GEF}, depends on the size of the aircraft:

$k_{GEF} = 9.1$ for commercial aircraft with $N_{pax} \geq 60$

$k_{GEF} = 15.2$ for business jet and commuter aircraft $10 \geq N_{pax} > 60$

$k_{GEF} = 25.3$ for smaller aircraft $N_{pax} < 10$

Although the above equations can also be used to estimate W_{GEF} for commercial aircraft, the current study focuses more on smaller aircraft; therefore, detailed validation was only performed for aircraft up to the size of an Airbus A320. It must be noted that larger variations can occur for galley weight depending on the equipment level for the various flight operations. However, the presented method is suitable for comparing different power system architectures for the same baseline aircraft.

Table 3-10 Galley, entertainment, furnishing.

Attribute	Parameter
External Parameters	
Geometry	Length of fuselage Width of fuselage Volume of cabin
Systems Architecture	Voltage Voltage type Number of subsystems Electrical system association
Aircraft	Number of engines Number of passengers Number of crew

Lights

The aircraft lights provide visibility and visual communication to ground units and aircraft. The subsystems include various exterior and interior lightings within the aircraft to ensure safe operation. The exterior lighting includes typically red and green navigation lights on the wingtips to identify the aircraft direction. The strobe lights include flashing high-intensity lights specifically designed to alert other aircraft and ground units about the presence of aircraft.

The landing lights are powerful forward-facing lights used during take-off, landing, and taxing flight phases. The light is designed to illuminate the runway and the surroundings in dark or low light conditions and provide visibility to the pilots. These are typically operated by the pilots while take-off and landing. Modern aircraft feature automated lighting. These high-intensity lights allow pilots to view the runway, obstacles, and signs.

The interior lighting includes the cockpit lighting for the aircraft instrumentation panel, controls, and switches. The ceiling and overhead lights are attached throughout the passenger cabin. The reading lights are focused lights provided for reading and similar activities for passengers. Additional lighting can be customizable for the business jet variant, which features additional power demand to increase customers' comfort level.

The current workflow adapts simplified lighting power demand estimation based on the length of the fuselage, $L_{fusleage}$. It is assumed that lengthening the fuselage requires additional lightings to illuminate the particular region added to the aircraft. The power demand for the lights can be estimated by Eq. (3-18)

$$P_{Lig,nom} = 0.31\ L_{fuselage} \tag{3-18}$$

However, it should be noted that the simplified estimation should be replaced by the physics-based method, which is more realistic to capture variations in light architecture. Moreover, the technology considerations require additional factors to the equation. The weight estimation of lighting is included within the galley, entertainment and furnishing as lights are considered to be an interface.

Avionics, Instruments

The avionics and instruments consist of navigation, communication, flight management and aircraft control subsystems. The flight management system (FMS) are navigation and guidance subsystem that assists pilots in route planning and flight management. Typically, these are responsible for displaying the optimized fuel-efficient flight routes. The navigation subsystems include the Global Positioning System (GPS), Inertial Navigation System (INS) and VHF omnidirectional Range (VOR). These subsystems are responsible for providing the pilots with positioning and route guidance and remaining route distance data.

Modern aircraft consist of autopilot systems responsible for automatic flight control actuation. The autopilot subsystems can maintain the aircraft's preset aircraft altitude and heading, making the automatic landing possible. The flight control computers (FCC) are an interface unit considered within the avionics subsystem in a conventional fly-by-wire aircraft. These redundant units provide the control signal to the FCS subsystem according to the pilot or autopilot inputs. However, modern electrified aircraft require subsystems such as Power Control Electronics (PCE), which are additional to the FCC. Hence, further studies are required to establish the subsystems within the aircraft systems category.

The pilots establish communication between the aircraft and the ground station with the communication subsystems. Typically, these are in the form of radio waves, Very High Frequency (VHF) and High Frequency (HF) and Aircraft Communications Addressing and Reporting Systems (ACARS) for sending and receiving text messages and data. Avionics subsystems include weather radar, which reports real-time hazardous weather conditions. The Traffic Collision Avoidance System (TCAS) alerts the pilots about the incoming nearby aircraft to avoid potential collisions.

The aircraft flight instruments include subsystems that give critical flight information to the pilots. Conventional aircraft include an analog display. However, modern aircraft include digital displays. The subsystems include airspeed indicators, altitude indicators, Vertical Speed Indicators (VSI), heading indicators, Turn coordinators and Horizontal Situation Indicators (HSI). The modern digital display integrates multiple analog display subsystems into a single screen and heads-up display to increase situational awareness and reduce instrument errors. Additionally, many customized subsystems can monitor engines and other flight-critical subsystems, depending on the aircraft generation and type.

These subsystems are responsible for safe operation and ease pilots for their functionality. This electronic equipment thus requires power input in normal and degraded operation. The current workflow estimates the overall electric power demand by the avionics subsystems using empirical correlation to aircraft fuselage length, $L_{fusleage}$ using Eq. (3-19)

$$P_{Av,inst_{taxi}} = 0.612\, e^{0.048\, L_{fuselage}} \qquad (3\text{-}19)$$

However, it should be noted that further physics-based models are required to carry out subsystem-level architecture analysis on avionics and Instruments. The weight estimation of the avionics and instruments is considered as per the NASA Flops method.

Table 3-11 Avionics, instruments input/outputs

Attribute	Parameter
External Parameters	
Geometry	Length of fuselage Width of fuselage

Systems Architecture	Voltage Voltage type Number of subsystems Electrical system association
Aircraft	Number of engines Number of crew Engine mounting Mach number

Air conditioning

The aircraft air conditioning unit consists of subsystems responsible for maintaining a comfortable environment inside the cabin. These consist of heating, ventilation, and air conditioning subsystems. Typically, these perform heating, ventilation, and air conditioning to regulate the passengers and crew members' temperature, humidity and air quality. The conventional aircraft features bleed air intake for the air conditioning system. This hot and non-conditioned air must be cooled and conditioned before it can be fed into the cabin. The air conditioning pack typically located in the fuselage functions to cool the incoming air from the engines by removing excess heat and moisture. In a more electric aircraft configuration, the bleed air intake is removed with a dedicated air intake scoop to increase the overall efficiency of the aircraft.

The temperature control and mixing subsystems regulate the cabin temperature, maintaining the desired comfortable temperature by varying the airflow inside the cabin. The subsystem also removes excess air moisture to prevent condensation-related problems. The aircraft is equipped with a humidification system to maintain the humidity level inside the cabin, typically between 10% and 20% [68]. Mixing the fresh air with the recirculated air increases efficiency by reducing the bleed air requirement. About 35% to 50% of the air is typically circulated inside the cabin [68].

The pressurization of the cabin is accomplished by the environmental control system whenever the aircraft is designed to fly more than 10,000 ft. The passengers are susceptible to discomfort or hazards at higher altitudes where the pressure difference compared to sea levels can be very high. There are further subsystems in addition to the air conditioning pack, such as pressure controllers and outflow valves to regulate the air entering and leaving to maintain the desired pressure level. A cabin altitude pressure level of 6000 to 8000 ft is typically maintained to ensure a comfortable environment.

The air filtration subsystems filter the incoming and recirculated air to remove particles, allergens, and contaminants. The High-Efficiency Particulate Air (HEPA) filter filters airborne particles, including bacteria and viruses. The subsystems can remove particles up to 0.3 micrometres with an efficiency of 99.97%. Typically, these are installed within the aircraft recirculation system, where the air is filtered before it is mixed with fresh air from the bleed or scoop. The filters contribute greatly towards maintainability since they must be replaced due to their short lifespan.

Conventional aircraft air conditioning units use bleed air from the engines. This corresponds to pneumatic power requirements in terms of bleed air flow rate requests to the power generation system. The pneumatic ducting required to carry the airflow is considered within the pneumatic system. However, the minimum electric power demand by other air conditioning systems can be estimated by the methodology established by Esdras and Liscouët-Hanke [67].

$$P_{ac,nom} = 0.077\, V_{cabin} - 0.40 \tag{3-20}$$

The bleed less architecture consists of an electric motor driving the compressors in a more electric configuration. The particular electric power demand request is demanded towards the power transformation and distribution system. The weight estimation of the air conditioning pack is considered as per the NASA Flops method.

Table 3-12 Air conditioning input/output

Attribute	Parameter
External Parameters	
Geometry	Volume of cabin
Systems Architecture	Voltage Voltage type Number of subsystems Electrical system association
Aircraft	Number of passengers Number of crew

Ice protection system

The aircraft ice protection system consists of subsystems responsible for preventing or removing the ice formation that accumulates on the critical surfaces that prevent the aircraft's safe operation. These are some of the prominent problems when the aircraft is operational during cold weather conditions or at the altitude of operation. The favourable temperature for the ice formation is between 0 to -20ºC. For a typical mission of aircraft operating at a very high altitude, take-off, approach, descent, and landing are the main flight phases affected by the ice formation. However, the ice formation can occur during the entire mission for smaller aircraft without pressurization, operating at a low altitude. The main focused components are the wing, tail and engines. These can affect the aircraft's aerodynamic performance and compromise flight safety.

The aircraft uses two types of ice protection systems: de-icing and anti-icing. Typically, the de-icing system is common in smaller variants and business jet aircraft, while the anti-icing system is common in medium-sized and large commercial aircraft. The anti-icing subsystems are designed to prevent ice formation on aircraft surfaces. The method used is to heat the affected surface to increase the temperature enough to prevent ice formation. The heating of the surface prevents the moisture buildup on the surface. In a conventional aircraft, the pneumatic bleed air is used to supply hot air through piccolo tubes to the surfaces. However, heating elements are installed in a more electric variant, which uses resistive heating of the surface. The electrical system supplies the power demand corresponding to the electric variants.

The de-icing system is quite different from the anti-icing system. The anti-ice system works by preventing ice formation. This means that the anti-icing system cannot immediately remove the ice whenever an ice build-up is already on the surface. The most common type of aircraft de-ice subsystem is the pneumatic boots. These boots are made of rubber-like material installed on the specific affected surface. In conventional aircraft, the boots are inflated with pneumatic air from engines, breaking the ice formation. Hot air also acts as a de-icer, although the reaction may not be instantaneous.

The current workflow accounts for power demand from the anti-icing system in terms of electric power demand. This is considered to be varying with respect to the wingspan, s_w of the aircraft, presented in Eq. (3-21) The electric power demand for the electrothermal de-icing can be considered to be 5% of $P_{ips,nom}$.

$$P_{ips,nom} = 0.035\, s_w + 2.02 \tag{3-21}$$

The weight of the anti-icing subsystems is estimated as a whole with the NASA Flops method [30]. Due to the extra inflatable rubber material, the pneumatic boots de-icers are much heavier than the anti-ice subsystems. This is three times heavier compared to conventional anti-ice systems.

Table 3-13 Ice protection system input/output

Attribute	**Parameter**
External Parameters	
Geometry	Wingspan Fuselage width Wing sweep Diameter of engines
Systems Architecture	Voltage Voltage type Number of subsystems Electrical system association Type of ice protection
Aircraft	Number of engines

Fuel System

The connection of the fuel system is established with the physics-based method, as explained in the previous section, 3.1.1. The weight estimation of the fuel system is done by using the object connections established by the Systems MDA tool. However, the electric power demand estimation is estimated within the Systems MDA module is sensitive toward the fuselage length., $L_{fuselage}$ and number of engines N_{eng} and is presented in equation.

$$P_{FS,nom} = 2.88\, e^{\frac{0.0399\, L_{fuselage}}{N_{eng}}} \tag{3-22}$$

3.4 Systems Safety Considerations

The Systems MDA module sizes and analyses safety-critical aircraft systems architecture. The system's safety remains a critical topic within the aviation industry to determine the feasibility of the architecture. Regulatory bodies such as the Federal Aviation Administration (FAA), and the European Union Aviation Safety Agency (EASA) require robust safety analysis as a part of certification compliance criteria for aircraft systems. The aircraft systems being components susceptible to failure, it is necessary to prove the reliability of the overall design remains compliant and is available during critical flight operating conditions such as single failure or dual failure, including engines. Hence, introducing newer technologies faces several challenges regarding safety compliance and may not be feasible for certification.

The Systems MDA module applies simplified safety considerations within the framework. The safety critical considerations are implemented within the physics-based modules. The flight control systems consist of subsystems such as actuators and power electronics, which are considered with utmost importance in terms of safety. The simplified safety approach applies all the actuators assigned to a specific control surface to the maximum hinge moment. Although during the conceptual design phase, this means that the FCS weight estimation could yield heavier weight, this ensures the architecture the safety rules and weight-based integration can be done to eliminate non-feasible architectures. The module is compatible with the safety rules presented by Jeyaraj et al. [22] which is to be integrated into the workflow.

The hydraulic system sizing depends on the connected hydraulic subsystems. The framework is built based on individual systems which can supply the flow. The safety considerations of subsystems such as FCS actuators connected to the hydraulic system are based on the hydraulic channels assigned to the control surface. The Systems MDA module assumes that the assigned hydraulic distribution architecture to the subsystems corresponds to a feasible set of architecture. The hydraulic generation architecture safety considerations correspond to a simplified approach based on SAE ARP [62] which corresponds to setting the EMP to compensate for the flow during the descent phase and prevent the EDP from being oversized. However, for a more electric configuration, eliminating EDP leads to EMP being able to supply maximum flow. The Systems MDA employs a flow ratio factor to be applied to share the flow with EDP and EMP in case both subsystems need to be set as main. However, setting the ratio requires additional integration with respect to the overall hydraulic and electrical system systems safety analysis based on Jeyaraj et al. ASSESS tool [22].

The electrical system sizing depends on the electrical consumers connected to the electrical system. The safety considerations for the subsystems connected to the electrical system are based on the assigned electrical distribution. Hence, the electrical framework assumes that the assigned architecture is feasible or controlled by the architecture descriptor. The electrical conversion is assumed to be equivalent to the number of distributions. For example, in case there is a 270 V DC feeder, it is assumed that it is interfaced with a power converter to convert the voltage from the generation voltage 115V AC. The simplified safety considerations for the electrical generation are adapted from [7]. The electrical system generation is applied with three scenarios: one generator fails, one engine fails, and one engine opposite generator fails. The electrical system within the workflow is assumed to supply the essential loads during all the flight critical scenarios. Moreover, Systems MDA assumes that all the electrical generations are available to supply the essential loads for all distribution and conversion due to the interconnection between them.

3.5 Systems MDA Mission Analysis

The aircraft systems mission analysis involves analyzing and evaluating variations in the subsystems performance and capabilities for a specific mission or operational criteria specified by Top Level Aircraft Requirement (TLAR). The mission analysis differs from the sizing analysis, where the worst scenario is sized to estimate overall weight variations in the architecture. The mission analysis aims to ensure the aircraft systems support the specified mission.

The main focus of the Systems MDA mission analysis is the evaluation of the interoperability and interaction of the aircraft systems during the mission. The analysis aims to examine the performance of the integrated system with respect to the timestep in communicating with each other and exchanging data to validate its functionality. For example, in mission analysis, output from each module, such as flight controls and hydraulic systems, is updated during each timestep to estimate the electric power for the electrical system. The net electric power demand corresponds to the power demand incurred for a specific mission.

The aircraft systems mission analysis requires the definition of two major criteria. The first one is the mission requirements, and the second is the objective of the analysis. The major requirements for the mission analysis are the mission requirements and the updated sizing results of the aircraft systems. The mission requirements are typically defined at Top Level Aircraft Requirements, an input to the Systems MDA module. The mission profile is used to set the timesteps and other operational criteria for the aircraft systems. Furthermore, novel subsystem integration leads to further detail in the connection detail regarding the data exchange between the subsystems during the mission operation.

The second criterion is the overall objective of the mission analysis, which sets the outcome of the mission analysis. The trade-off between the subsystems is set with respect to the objective of the mission analysis. Conventional objectives in the aviation industry focused mainly on the weight reduction of the aircraft. A major focus was given to the overall systems weight reduction, giving sizing analysis as the main criteria for selecting systems. However, today's focus on reducing carbon footprint has shifted the objective towards reducing fuel burn.

The Systems MDA mission analysis estimates the power requirement and concurrent fuel consumption due to the connected subsystems. *Systems MDA* estimates net mission fuel consumption due to systems as per SAE AIR1168/8[69]. Conventional aircraft feature engines as the biggest fuel consumers. More recent aircraft configurations, such as hybrid and all-electric variants, reduce fuel consumption by powering the propellers using the electric motor. The aircraft systems feature electric generators and hydraulic pumps which are connected to the engines via engine gearbox and shaft power, extracting power from engines. This power is translated in the form of mechanical power demand towards the engine, which acts as a fuel weight increment due to aircraft systems. The net shaft power demand can be estimated from Eq. (3-5) and Eq. (3-15)as follows:

$$P_{shaft,eng} \ = \ P_{shaft,hyd} + P_{shaft,elec} \tag{3-23}$$

where $P_{shaft,eng}$ is the net engine shaft power.

The total fuel consumption is due to the fixed systems weight component itself, shaft power demand, bleed air component, ram air component and variable weight penalty, which corresponds to expandable material.

$$W_{F,sys} \ = \ W_{F,weight} + W_{F,bleed} + W_{F,shaft} + W_{F,ram} + W_{F,varweight} \tag{3-24}$$

Scholz [70] provides a methodology with which to estimate the specific fuel consumption due to systems (SFC_p). According to Scholz, a shaft power factor (k_p) translates SFC into SFC_p. As per the turboprop engine used on the DO-228, and data availability [70] , a k_p value of 0.00404 is assumed in the current workflow for the fuel weight estimation. The Federal Aviation Regulation (FAR) Part 23.831 and Part 25.831 set the bleed air requirement as 0.55 lb/min/pax. This is translated into the bleed air requirement for conventional configurations. MEA and AEA configurations without engine bleed off-takes have dedicated ram air inlets, and the bleed air requirements are translated into ram air requirements.

The more electric and all-electric aircraft features novel subsystems and focuses on the reduction of fuel consumption. The addition of novel subsystems such as batteries dedicated towards secondary power consumption and solar power system (SPS) features additional challenges towards the mission analysis. In addition to the sizing of the system, Systems MDA features transient analysis, which estimates the power demand with respect to the timestep.

3.5.1 Integration of Novel Disciplines

The Systems MDA tool is required to have the capability to integrate novel subsystems to conduct aircraft-level analysis of novel architecture. This requires the capability to estimate the power demand with respect to the required time step in line with the aircraft sizing or the connecting subsystem. The requirement can be defined at the TLAR or specific system level as per the MDA requirements. The Systems MDA considers a simplified approach to estimate the transient power demand of the subsystem. Each subsystem is considered to consume maximum power demand during the peak power demand and nominal power demand during the normal operating conditions. The power demand is considered to be a normal distribution with respect to the power off-take during the specific flight phase. Subsystems can be configured individually to have specific standard deviations depending on the known values or data availability. For the purpose of simplicity, it is assumed that the standard deviation is about twenty percent by default. Moreover, individual subsystems' overall power demand is considered active-active preset. Apart from this, the parameter power on time and power off time of individual subsystems are considered based on percentage with respect to the operational mission time. For example, a twenty percent value would yield the subsystems drawing maximum power within the flight phase for twenty percent, while the rest consume nominal power. Since the analysis estimates the power demand at individual time steps, the analysis requires the sizing analysis as an input to be completed.

The Systems MDA power generation system consists of engines as the primary power source for the power transformation and distribution system. The more electric and all-electric version focuses on the electrification of the unit, thereby exploring the reduction of fuel required for the mission. The Systems MDA, power generation system estimates the fuel consumption based on the shaft power demand by the subsystems as discussed earlier. The Systems MDA mission analysis considers the power available from the novel subsystems to estimate the mission fuel consumption. The novel subsystems can be a Solar Power System or a dedicated energy storage module for the functioning of the systems. Moreover, the weight is considered in the sizing process as per the architecture inputs. The net shaft power demand can be estimated from total power demand and available power demand at each interval of timestep.

$$P_{net,shaft} = P_{total} - P_{available} \tag{3-25}$$

For example, the net power available from the Solar Power System (SPS) is used to update the shaft power demand required from the engines. This is used to estimate the reduction in fuel burn. In the cases where the total available power equals the required power, the net fuel consumption will be zero.

3.6 Implementation of the Framework

The overall framework implementation involves the choice of the platform which will enable integration and interaction between multi-tools chains. The primary step is to establish communication between the tools. The communication is established by exchanging the modifiers or manipulated parameters, which other modules can use. For example, the framework considers the output of the Aircraft sizing tool as an input for several tools, such as the Geometry and Energy storage tool. Yet another aspect of the platform should be the tool's or parameters' reusability. For example, the Geometry tool is required to be modified by the Energy storage tool, and the updated results are to be used for further analysis by the Fuel System tool. This way, the Geometry tool parameters are reused at every step in the MDA framework. The framework platform must also consider the addition of novel subsystems such as Systems Safety or Solar Power System (SPS) connectivity. The current framework focuses on building a sandbox platform to enable overall tool integration within the Top-level aircraft design environment. Hence, it is required to look at the possibility of the tool connectivity over a remote platform that can improve the tool results. This section will discuss the tool implementation platform and overall connectivity.

3.6.1 Object-Oriented Implementation of MDA Framework

The current framework is implemented using Object-oriented programming (OOP) approach. Efficient communication and coordination among the disciplines are crucial for the MDA framework. The OOP uses object-to-object interaction to manipulate data and methods within the objects, thereby effectively enabling the integration. The current framework tools, such as Aircraft Sizing, Energy Storage and Systems MDA, feature the OOP approach. Multiple instances can be created, which can later be used with multi-architecture inputs for further trade-off analysis.

The Systems MDA consists of several modules and submodules, such as a power-consuming system, power transformation and distribution system, energy storage and power generation system. These modules are considered as classes to enable object generation. However, for a single aircraft systems architecture, there will only be one object respectively. The submodules, such as flight control and hydraulic systems, are also considered classes to enable architecture-based object generation. The physics-based modelling allows the integration of subsystems such as actuators to flight control systems and generators into electrical systems. To enable this, all the main classes consist of several levels of inner classes. For example, the actuator is an inner class of flight control system. Further, the hydraulic system subdivision, such as distribution, generation and conversion, are considered inner classes of hydraulic systems, and subsystems, such as tubing, reservoir, etc., are considered inner classes of distribution.

The OOP classes or objects consist of their own data and methods. The data is treated as parameters, and the methods are the functions which are used to manipulate the parameters. The current framework is built based on a power flow pattern. Hence, the major estimation methods most common to all the classes and inner classes are power demand and weight. For example, the class flight control system, hydraulic system, and inner classes actuator have common methods electric power demand, hydraulic flow demand and weight. However, the data and model considered are different depending on the objects.

The requirement of the system object is the subsystem integration as an input. For example, actuator object generation is only possible towards specified control surface system objects. The object is linked in transferring the control surface data, such as wing area and control surface area data to the actuator object. The overall architecture determines the number of instances of objects. In this case, the total number of control surfaces and actuators within the control surfaces determines the object generation. This way, the instance mimics the real-world actuator object. This makes the subsystems easier to manipulate and to assign further properties such as volume and dimensions. Although the high-level conceptual design focuses mainly on high-level parameters, component-based integration, if required, can be further integrated with additional inner-class objects.

Consumer outputs are available to generate objects of further connected levels, such as power transformation and distribution systems, power generation systems, etc. For this purpose, subsystems are assigned to specific objects based on architecture. For example, specific subsystems such as flight control actuators or electric motor pumps assigned to specific electrical systems are only connected to the electrical distribution as per architecture. The final update output method within the classes and inner classes established the overall system subsystem integration by assigning the number of instances and outputs relevant to the overall design environment.

3.6.2 Data Exchange and Handling

The multidisciplinary tool interactions require an interface platform to enable data communication. The current workflow encompasses a Common Parametric Aircraft Configuration Schema (CPACS) [71] and an Excel-based platform to exchange and synchronize data between disciplines. This ensures that the common design variables shared between multiple disciplines consistently lead to a more efficient MDA analysis. The section will explore more detail on the data exchanges within the workflow.

The input variables within the workflow are categorized based on purpose and nature. The most used categories are aircraft parameters, geometry, and systems architecture. The aircraft parameters include TLAR and mission-level parameters such as Mach number, number of engines, etc. The geometry parameters typically represent the dimensions and shape of aircraft and systems, such as fuselage length, control surface area, etc. The systems architecture includes input representing system-level parameters, including technology and representation of the physical system. For example, generator voltage, voltage type, pressure, etc.

The CPACS-based platform allows interconnection between the design variables, keeping the disciplines independent. This means all disciplines communicate to the common platform to exchange data. Figure 3-14 shows an example of a CPACS file showcasing the tool structuring.

```
<toolspecific>
  <ASCON>
    <Converger>
    <Miscinput>
    <Aircraft_Sizing>
    <Geometry>
    <ASSET>
      <Inputs>
      <Energy_Storage>
      <SPS>
      <FCS uID="FCS_1">
      <Galley_Furnishing uID="Galley_Entertainment_Furnishing_1">
      <Lights uID="Lights_1">
      <Fuel uID="Fuel_1">
      <Avionics_Instruments uID="Avionics_Instruments_1">
      <AirConditioning uID="AirConditioning_1">
      <IPS uID="Ice_Protection_1">
      <Landing_gear uID="Landing_gear_1">
      <Hydraulic_System uID="Hydraulic_Systems_1">
      <Electrical_System uID="Electrical_Systems_1">
      <Misc uID="Misc_1">
      <Outputs>
```

Figure 3-14 Overall tool structuring

The input-output module corresponding to the disciplines facilitates the variables' propagation to and from the CPACS file. The sequence of the execution is responsible for the propagation of the variables. For example, the Aircraft Sizing tool updates energy storage power and energy requirements. The corresponding variables should be written or updated within the CPACS file before executing the Energy storage module. Similarly, the Energy Storage updates the space requirement to the geometry parameters, after which the fuel system estimates accurate sizing.

The Systems MDA module consists of external and internal variable updating. The external updating consists of overall parameter updating into the CPACS file relevant to the top-level aircraft analysis. For example, overall systems weight and power are considered as external variables. The internal variable updating consists of methods defined to update the subsystems towards the systems. For example, once the generator objects are updated with the electrical generation, the total number of generators is updated with the total number of objects connected to the generation.

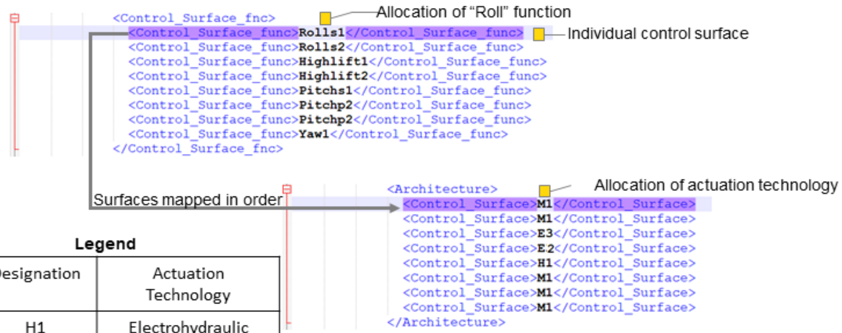
The output variables are quite similar to the input variables. Each discipline is considered to have a specific set of design variables written to the CPACS. These are generally written towards specific output categories in the platform. However, certain interface variables may serve as input variables for subsequent analyses in other disciplines. The current workflow enables the interface parameter propagation from the output analysis of each discipline as per the sequence to avoid parameter duplication. However, building an additional tool interface platform to update each discipline's parameters is also possible. The capability will be dependent purely on the overall MDA driver requirement.

3.6.3 Architecture Description

The aircraft systems analysis requires several architectures to be sized and analyzed at the system and subsystem level to analyze the feasibility of the objective. This also should ideally involve several parametric analyses or design of experiments to verify the tool interfaces with respect to real-time variations with other disciplines. The architecture descriptor within the Systems MDA tool drives the architectural changes, enabling architecture-based trade-off analysis. The architecture descriptor contains information about the subsystem and component level architecture, including power allocation and control allocation for each system architecture element.

The *Systems MDA* module in the presented framework employs a modified CPACS-based descriptor. This simplified descriptor stores aircraft-level parameters in a standard CPACS file but captures system-level parameters and the systems architecture using custom tags defined within the tool-specific tag of CPACS.

The simplified architecture descriptor in the Systems MDA can store information at multiple levels of granularity. For example, physics-based modules such as flight control, hydraulic, and electrical systems store subsystem and component level information. For example, the flight control system requires allocation of control surface definitions such as aileron, spoiler and specific functions to be assigned such as roll or auto brake, and technology information regarding the type of actuator such as EHA or EMA. Moreover, the assignments include allocating each consumer subsystem towards the hydraulic or electrical system depending on the power characteristics, such as one or left electrical system. An example of control surface assignment, function and actuator assignment within Systems MDA is shown in Figure 3-15.



Legend

Designation	Actuation Technology
H1	Electrohydraulic Servo Actuator
H2	Hydraulic Power Drive Unit
H3	Horizontal Stabilizer Actuator
E1	Electro-hydrostatic Actuator
E2	Electro-Mechanical Actuator
E3	Electric Flap Drive
E4	Electric Backup Hydraulic Actuator
M1	Mechanical Linkage

Figure 3-15 Systems MDA control surface definition and assignment of functions and actuators for an FCS

The hydraulic system includes a description for allocation of distribution, conversion, and generation subsystems within the system. Each of the hydraulic system architecture includes tubing, hydraulic pumps, reservoir allocation and type of hydraulic fluid details. Consumers are allocated toward the hydraulic system. Moreover, the hydraulic system allocation to the engine detail is added to the hydraulic system descriptor. A specific tag assignment within the system can assign the hydraulic system as a main or backup.

The electrical system description is kept like the hydraulic system. Assignments such as left or right electrical systems are based on the main electrical system tag with a specific uid. The uid is further used to allocate consumer subsystems to the electrical system. Electrical subsystems such as feeders, power converters, and generators are assigned voltage levels and architecture variations. The interconnection between the electrical system is taken care of automatically by the Systems MDA module. The electrical system can also be assigned to a specific engine as a main system or backup system as per the architecture.

The description of other consuming systems, such as galley, entertainment, furnishing, lights, avionics, etc., are not physics-based. Hence, component and subsystem level weights are assumed to be shared within the subsystem allocation. Simplified assignments such as voltage level and voltage type assignments are employed to enable the electrical and hydraulic system estimation.

The Systems MDA descriptor serves as a simplified descriptor to enable the parametric analysis with respect to the selected minimum set of feasible architecture. More sophisticated descriptors are to be used to increase the number of architectures studied to enable full-scale MDA analysis. The *Systems MDA* descriptor is compatible with the graph-based architecture descriptor of Jeyaraj et al. [22] (developed to allow further safety analyses), such that the information stored in the graph-based descriptor can be written directly into the *Systems MDA* descriptor file. Further examples and illustrations of the *Systems MDA* descriptor are shown in Figure A-1to Figure A-4 in Appendix A.

3.6.4 Integration with Collaborative MDAO Environment

The current workflow explores the feasibility of being part of the top-level aircraft design framework, such as the AGILE 4.0 project. To enable this, the tool must be feasible with remote-based distributed environments. Remote Component Environment (RCE) is a software framework developed by the German Aerospace Centre (DLR) to enable collaborative and distributed aircraft design modelling and analysis [52]. RCE allows aircraft design experts to generate workflow by integrating several disciplinary modules to enable efficient data processing and analysis.

The graphical user interface (GUI) within RCE allows design experts to visually create a workflow by connecting the disciplines. All the individual disciplines can be built as a package, making the integration simpler. Moreover, RCE allows multi-programming language tool integration, enhancing the feasibility of matured expert discipline within the workflow.

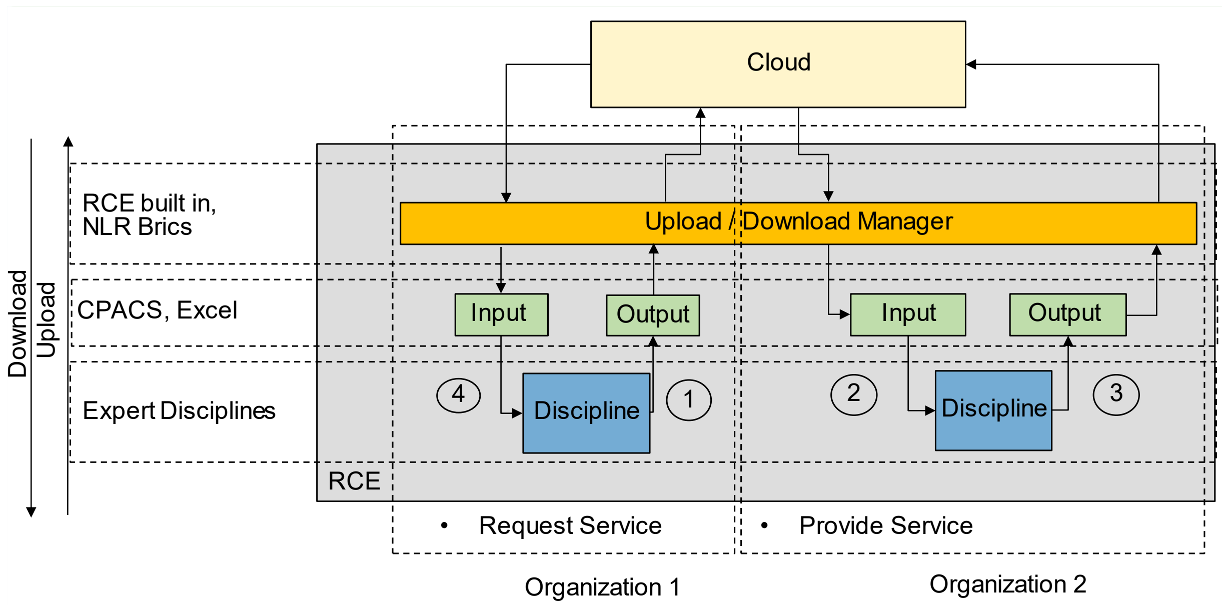


Figure 3-16 Remote-based workflow integration

The RCE enables user collaboration by providing packages for sharing and reusing the workflow and components. The remote component allows the sequence execution of the disciplines, which can be executed upon request by the integrator.

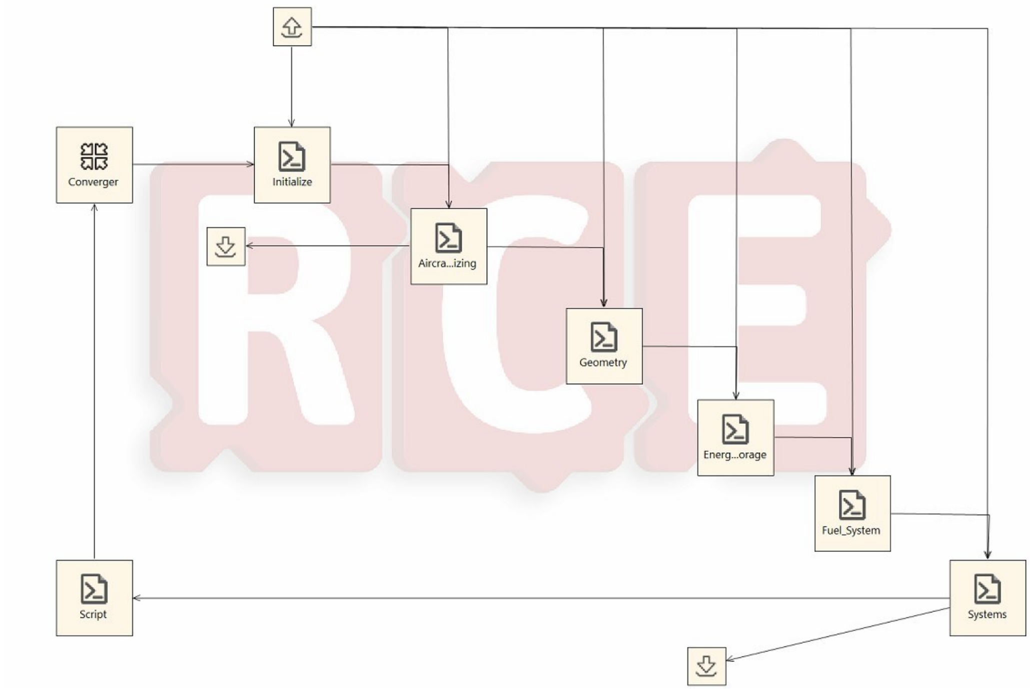


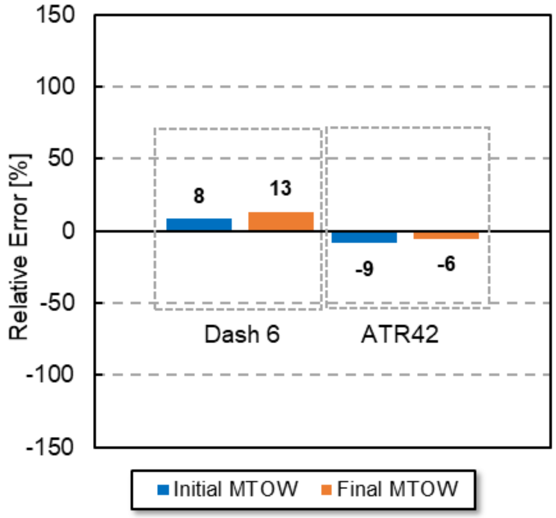
Figure 3-17 RCE tool setup

The current workflow is feasible within the RCE component. Figure 3-17 shows the overall tool layout within RCE. Each discipline, such as Aircraft sizing, Energy storage, Systems MDA, etc., is independently built as several packages. The programming language for all the tools remains Python, except the geometry and fuel system tools, which are Excel-based. However, the Python-based interface program controls the read and write capability of the tools. All the disciplines are connected in a sequence of execution. The interface parameters which are exchanged are defined within the RCE interface, which takes care of reading and writing while executing. All the other parameters are controlled by individual discipline.

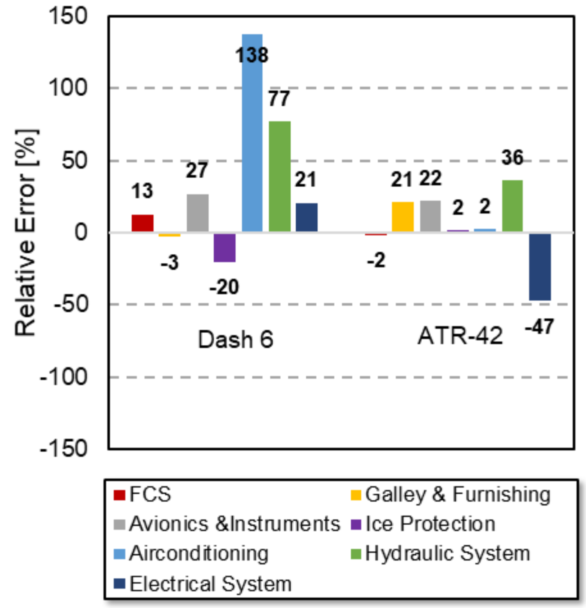
The remote component exchanges the CPACS file with relevant inputs as a request service. The individual disciplinary experts are responsible for executing the discipline. The end product is the same updated CPACS file with the relevant output parameters.

3.7 Tool validation

The subsystem tools implemented in the workflow are validated using data available in the literature. The detailed subsystem validation results are presented in the corresponding previous sections. The example of the overall workflow validation of the results for two selected aircraft in Figure 3-18 below. The Dash 6 and ATR42 have been selected as they are conventional turboprop aircraft in the regional and commuter aircraft category that is of interest in this case study. Only very few validation data are publicly available for aircraft subsystem weights in this aircraft category. The data for the validation cases were extracted from [72,73] and other public sources.



a) MTOW validation



b) Subsystems validation

Figure 3-18 Overall tool validation

Figure 3-18 a) presents the results for overall MTOW error variation. The aircraft-level validation is performed with respect to the overall takeoff weight of the aircraft. The initial MTOW corresponds to the initial weight from the aircraft sizing tool. This corresponds to aircraft sized using conventional methods without detailed subsystem sizing estimations. The final MTOW corresponds to updated weight estimation adding delta weight variations from multiple disciplines. The results show an error variation within -9% to 13%, which is good for conceptual design methods.

Figure 3-18 b) shows the individual subsystem sizing error variation in terms of weight estimation. The individual subsystem error variation is larger which can be attributed to variations in the subsystem analysis methods and inputs. As mentioned in the previous subsections, the subsystems which follows empirical, or NASA Flops method follows correlations contributing to error propagation. The subsystems which follow physics based method consist of majorly input and model based errors. For example, the input derived from public sources such as P/W ratio of component or shaft power factor (k_p) of engines etc. contributes towards errors. Moreover, due to unavailability of inputs in the early design stage there are several simplifications applied within the models, such as fixed bundle loading for wiring, simplified routing estimation without considering clashes, which contributes to additional errors. However, as the weight of the overall system is acceptable (in the range of 10%), and since we observe the correct trends for architecture and technology options, we consider the tool performance suitable for the presented case study.

However, combining various subsystem estimations with uncertainty and error variations calls for proper uncertainty management and analysis, which will be addressed in future work.

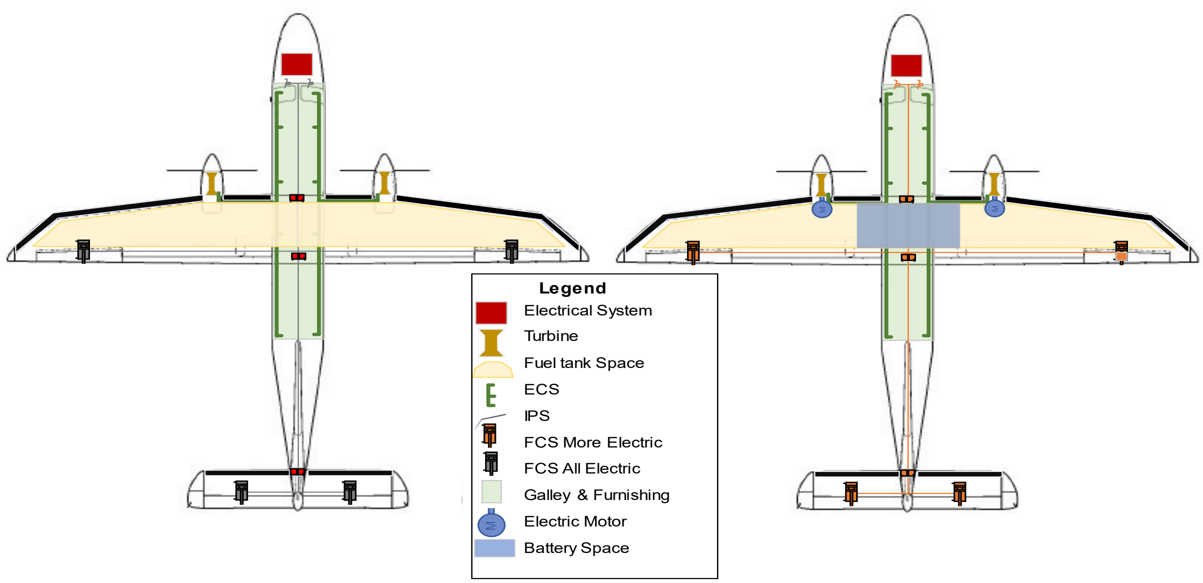
4 Results and Discussion

This chapter shows the application of the framework with several case studies on hybrid electric aircraft.

4.1 Case Study for the Electrification of a Commuter Aircraft

The overall methodology is applied to carry out an analysis of the hybridization of DO-228 aircraft. The aircraft is chosen due to the high data availability of aircraft and subsystems. The DO228 is a twin-turboprop high-wing aircraft designed and manufactured by a German aerospace company. The aircraft consists of variants such as commuter or utility and is considered multi-role. The conventional DO-228 consists of turboprop engines, while the current case study explores the possibility of hybridization using parallel hybridization. Architecture consideration includes additional energy storage (battery) to run the propulsion electric motor. Figure 4-1a shows the conventional DO-228 configuration, whereas Figure 4-1b shows the hybrid-electric version considered for the case study.

The conventional DO-228 is designed to meet the certification requirements of FAA Part 23. The aircraft can carry up to nineteen passengers and two to three crew members. The aircraft is designed and tested to function in all day and night weather conditions within the temperature limits of −40°C to +50°C. The aircraft is equipped with Garrett TPE-331-5-252D engines with an engine shaft power of 715 SHP. The aircraft uses engine bleed and heat exchangers within the air conditioning system to maintain the cabin temperature between +18ºC and +28ºC. The aircraft consists of de-icing boots and a pneumatic anti-ice system for leading-edge slats, propellers and pitot tubes. Two integral fuel tanks are located on each wing between the front spar and rear spar. The cross-feed allows the interconnectivity between the fuel tanks. The Fuel System uses jet pumps for interconnectivity. The hydraulic system powers the landing gear, brakes and nose wheel steering. The hydraulic power is generated with a standard DC electric motor pressurizing the system up to 3000 psi.



a) Conventional Do-228 b) Hybrid-electric configuration

Figure 4-1 Conventional and Hybrid-Electric DO-228 configuration

The electrical system of the conventional DO-228 is a 28 V single-wire installation, with the airframe used as a ground return. The system consists of two engine-driven generators, two batteries for the DC power supply, and two inverters for AC power consumers. All the inputs required for the analysis are summarized in Table 4-1.

Table 4-1 Key aircraft-level parameters for the DO-228 [72]

Aircraft Parameter	**Value**	**Aircraft Parameter**	**Value**
Wing span $[\mathrm{m}]$	16.97	Number of passengers	19
Aspect ratio	9.0	Hydraulic system pressure $[\mathrm{psi}]$	3000
Length $[\mathrm{m}]$	16.56	Generation voltage $[\mathrm{V}]$	28
Cabin volume $[\mathrm{m}^3]$	14.7	Generation voltage type	DC
MTOW $[\mathrm{kg}]$	6400	Inverter voltage $[\mathrm{V}]$	115
Propeller diameter $[\mathrm{m}]$	2.73	Wing area $[\mathrm{m}^2]$	32

The overall MDA tool validation is carried out with the conventional DO-228 architecture. The actual DO-228 MTOW from the references is approximately 6400 kg. The MDA final output is about 6372 kg. This corresponds to a −0.5% error variation.

4.2 Case study results

This subsection presents the results of the analysis with the complete workflow presented in the previous section. The aircraft-level input data required for the hybrid-electric case studies come from the literature [74]. The workflow considers a short-range mission, which is defined at a range of 213 NM and a payload of 1960 kg, to estimate the fuel burn for the hybrid-electric version of the aircraft. It should be noted that the range and payload remain the same for all comparisons. Additional structural reinforcement of landing gear is not considered directly to accommodate the increased aircraft MTOW.

Table 4-2 shows the results of the complete MDA for conventional (non-hybridized) DO-228 and the hybrid electric versions. The propulsion system architecture comprises a parallel-hybrid configuration with a hybridization factor of Hp = 0.1. The specific energy of the battery E_{sp} is assumed to be 272 Wh/kg (representing current battery technology from [75]), and futuristic case of 800 Wh/kg is chosen. Different system architectures are investigated, focusing on the various levels of electrification of the actuation systems and different voltage levels for the electrical system. All the key inputs and systems architecture variations are listed in Table 5-5 of Appendix B.

The current workflow analyses the overall fuel volume from the total fuel tank volume availability. However, for a hybrid electric variant where the battery space trade-off is carried out, the comparison needs to consider the reduction in the existing fuel tank volume. The geometric modeller estimates the overall fuel tank volume based on the layout, and the initial space assignment of allocation is assigned to the battery sizing tool. The updated space consumed by the battery from the space constraints and the aircraft level inputs are used to update the overall space availability. The remaining volume in terms of space availability is assigned to the *Fuel System* and is also an input to the *Fuel System* weight is sizing process. The finalized geometry and *Fuel System* weight is used within the *Systems MDA* framework to analyse the weight and power consumption due to all other systems.

Table 4-2 Results of the complete MDA for the baseline aircraft and the hybrid-electric version.

Framework	Aircraft Parameter	Value		
	Aircraft Configuration	Conventional	$H_P = 0.1$, AEA High Voltage	
			$E_{sp} = 272$ Wh/kg	$E_{sp} = 800$ Wh/kg
Aircraft Sizing	**Initial aircraft Sizing results**			
	MTOW [kg]	6378	9286	7139
	Systems Weight [kg]	1140	1140	1140
	Battery Weight [kg]	0	613.5	208.5
	Wing Area [m^2]	31	45	35
Aircraft Geometry	**Initial Battery Space Assignment**			
	Battery Assignment [x, y, z] [m]	1.42, 5.7, 0.14	1.42, 5.7, 0.14	1.42, 5.7, 0.14
Battery	**Final Battery Space Assignment**			
	Battery Final Space [x, y, z] [m]	0.13, 0.035, 0.082	0.7, 3.9, 0.14	0.78, 1.3, 0.13
	Battery Weight [kg]	0.61	636.4	221.4
	Battery Volume [m^3]	0.0004	0.38	0.028
Fuel System	**Fuel System Sizing**			
	Fuel Tank Volume [m^3]	2.16	1.77	2.13
	Fuel System Weight [kg]	79.8	76.6	78.8
Systems MDA	**Subsystems Sizing**			
	Systems Weight [kg]	1136	1213	1203
Aircraft MDA	**Final Integration results**			
	Wing Area [m^2]	31	46	35
	Fuel Weight [kg]	501	441	425
	OWE [kg]	4000	6800	4800
	MTOW [kg]	6373	9483	7234

The final integration uses the overall delta weight estimated between the initial *Aircraft Sizing* and the weight values calculated by the *Energy Storage*, *Fuel System*, and the *Systems MDA* tools. This weight value is used to update the OWE and, thus, MTOW. Table 4-2 shows the final results with the updated wing area due to the aircraft's overall weight increment. The systems and batteries remain the main contributors to the weight variation for a hybrid configuration. While the battery weight increment is due to hybridization, modifying the aircraft geometry to support new battery weight for a conventional systems configuration leads to an increment in systems weight.

4.2.1 Parametric analysis

This section presents a systems-specific parametric analysis carried out on DO-228 aircraft. The conventional aircraft configuration is electrified by modifying key aircraft systems to analyze the effect at the aircraft level. The study includes multiple hybridization factors such as 0, 0.1, 0.15, and 0.2 with the current battery technology (272 Wh/kg) to obtain more realistic system-specific effects on MTOW and the wing area.

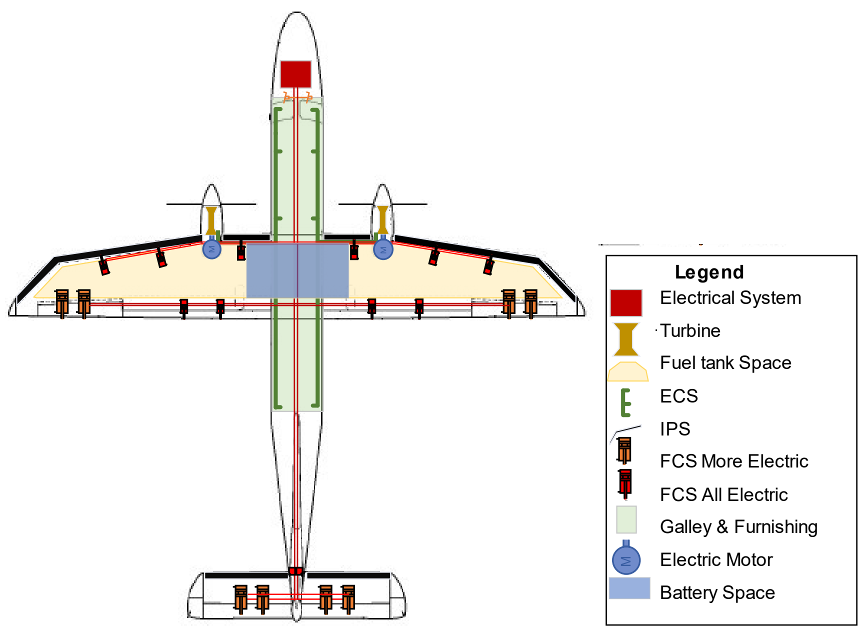


Figure 4-2 Electrified DO-228

Table 4-3 Systems architectures and technology configurations considered for the parametric analysis.

Parameter	Conventional	More Electric	All Electric
FCS primary flight control actuator	Mechanical	EHA, EHSA	All EMA [1]
FCS secondary flight control actuator	Mechanical	All EMA	All EMA
Power converter	DCAC, DCDC	ACDC, ACAC	ACDC, ACAC
Voltage level	Constant	Low	Medium High

FCS voltage (V)	28	28	270	270
FCS voltage type	DC	DC	DC	DC
Avionics voltage (V)	28	28	270	270
FCS voltage type	DC	DC	DC	DC
Other systems voltage (V)	115	115	115	230
Other systems voltage type	AC	AC	AC	AC
Electrical generation voltage (V)	28	115	230	230
Electrical generation voltage type	DC	AC	AC	AC

Figure 4-3 shows the impact of the electrification of the various subsystems on the system weight. The electrification of the aircraft leads to a systems weight increment compared to the conventional aircraft configuration.

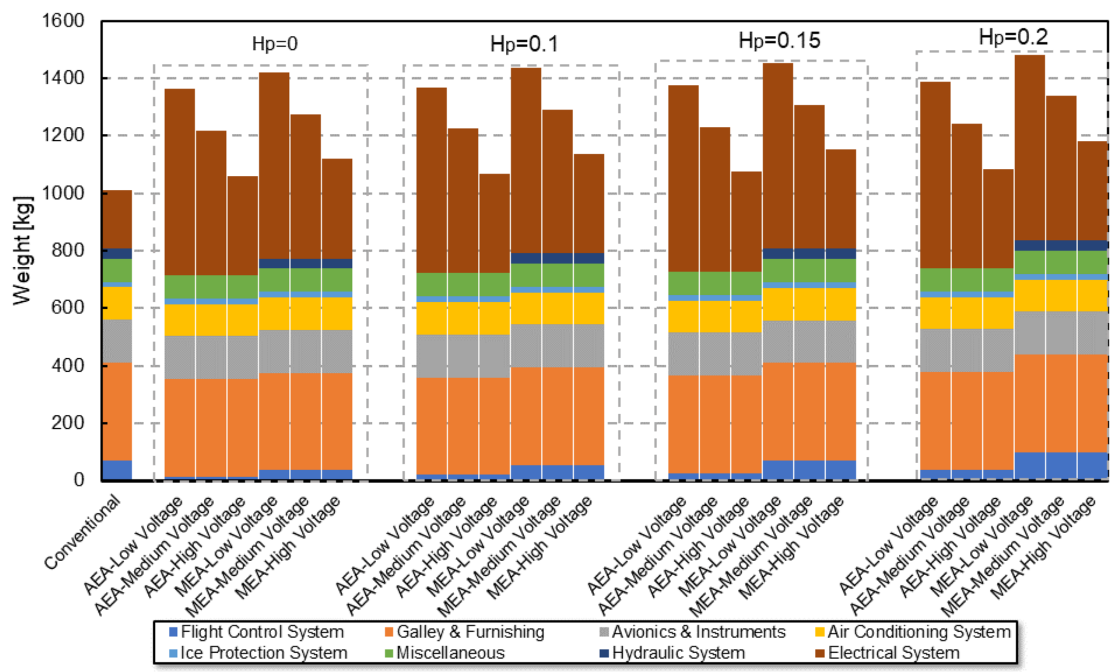


Figure 4-3 Electrification impact on subsystem weight for H_P, system architectures, and voltage levels.

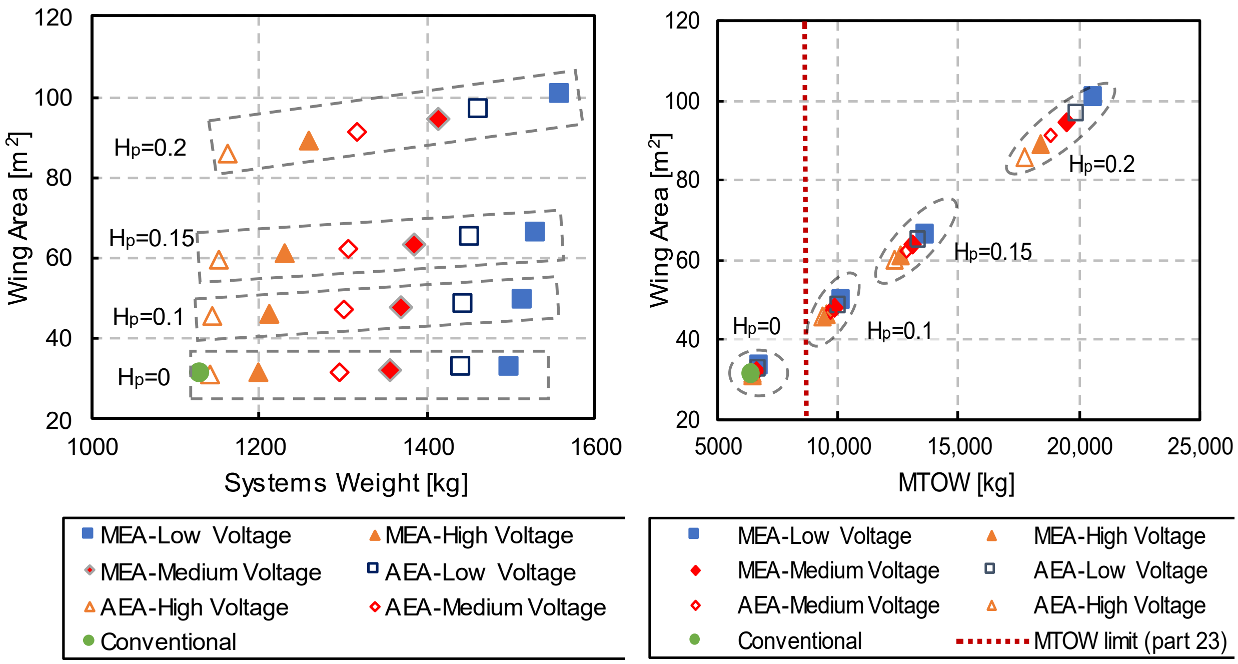
Compared to the conventional configuration, the FCS weight reduction due to the elimination of the mechanical linkages is seen for the MEA and AEA architecture. The much heavier version of pneumatic boots is removed, saving significant weight in the IPS. However, the net increase in electrical power demand due to electrification and added wiring significantly increases the electrical system weight. The electrified architectures with low-voltage power-consuming systems suffer from increased losses and require heavier feeders. Conversely, increasing the voltage levels leads to lighter feeders and reduces the overall weight of the electrical system. Within AEA and MEA architectures, FCS weight reduction occurs in AEA due to the application of EMAs, which are much lighter than other actuator types. Eliminating the centralized hydraulic system leads to a significant decrease in weight for the AEA architecture. Hence, a high-voltage AEA architecture is seen to be highly beneficial in a hybridized version of the aircraft.

Figure 4-4 summarizes the parametric study results focusing on the variations in wing area and weight for today's battery technology (E_{sp} = 272 Wh/kg [76]). Figure 4-4a shows the effect of the various parameter changes on the overall systems' weight (output of *Systems MDA* module). Figure 4-4b analyzes the impact of the same parameters on the MTOW. The overall MTOW increment due to hybridization is primarily due to the higher battery weight, which requires additional wing area to support the weight. The resulting increase in wing area, in addition to the electrification, leads to additional weight due to the systems. Furthermore, the rapid rise in MTOW between different hybridization levels is attributed to the so-called snowball effect, wherein the weight contribution of batteries leads to more power being required to carry out the mission, which further increases the aircraft's empty weight and, therefore, the maximum takeoff weight. The large MTOWs of the hybrid-electric configuration exceed the weight limit for Part 23 certification. Furthermore, from a retrofitting perspective, a reduction in passengers and payload would be required to make these configurations feasible. Although it may be possible to envision retrofitting the aircraft with a larger wing, the additional weight will require strengthening the airframe structure, which carries a further weight penalty. These aspects are not explored within the scope of this study.

One can observe that for varying hybridization factors, the contribution of the system architecture follows the same trend: the MEA low-voltage architecture (solid blue square) is the heaviest architecture in each set, requiring the largest increase in wing area. On the other hand, AEA and MEA architectures with high-voltage architecture (orange triangles) seem to provide the lowest weight and required wing area increase.

Comparing the MEA and AEA configurations at low and high voltage levels shows that higher voltages lead to lower overall system weight; this effect is also observable in the reduced required wing area. Furthermore, this effect is more pronounced at higher hybridization factors, implying that using higher voltage levels at a higher hybridization factor is worthwhile.

However, to understand the overall impact of electrification on aircraft performance, the fuel burn change needs to be investigated, as discussed in Figure 4-6.



(**a**) (**b**)

Figure 4-4 Impact analysis of the hybridization factor, systems architecture, and voltage level E_{sp} = 272 Wh/kg. (**a**) Variation of system weight with wing area and hybridization factor. (**b**) Variation of MTOW with wing area and hybridization factor

Figure 4-5 investigates the effect of varying battery energy density E_{sp} in combination with the system electrification at aircraft-level parameters for a fixed hybridization factor (H_{p} = 0.1). Figure 4-5a, b shows that increasing the battery energy density E_{sp} helps to reduce the OWE increase (compared to a conventional system architecture) and thus helps the required increase in wing area to limit the increase in MTOW. However, even with very high E_{sp}, the modified aircraft still requires a larger wing and has a higher MTOW than the conventional configuration.

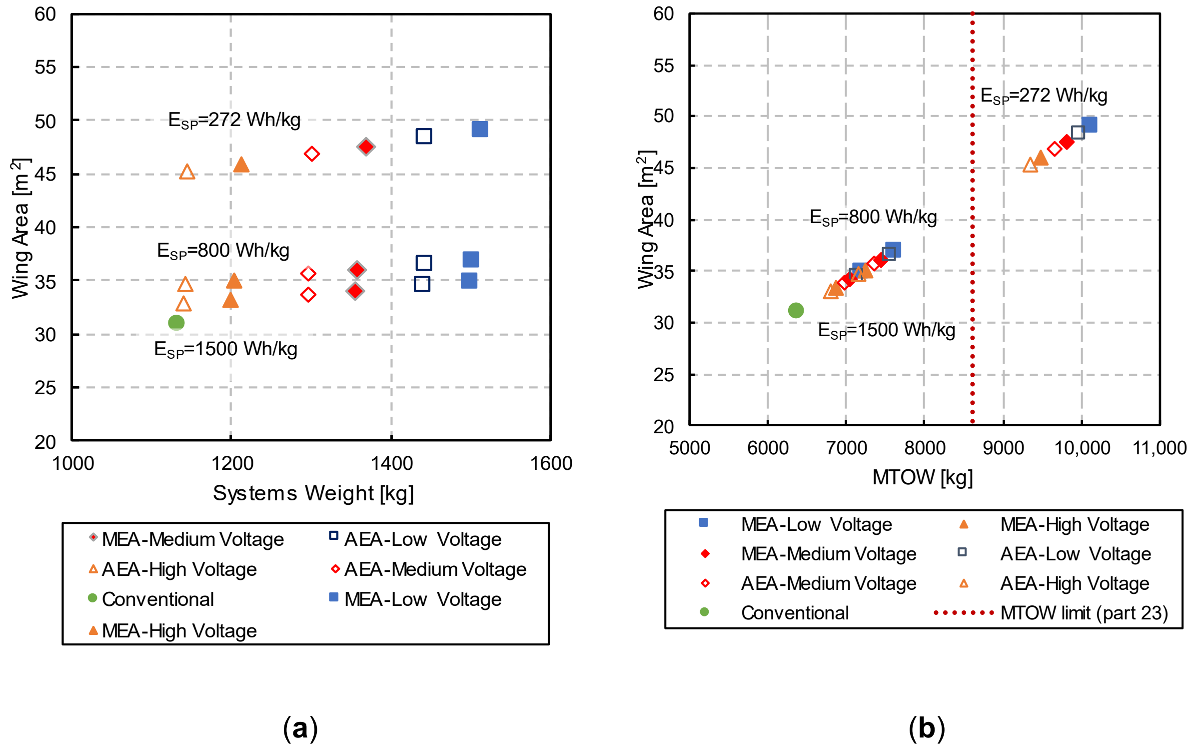


Figure 4-5 Impact analysis of the specific energy density, systems architecture, and voltage level Hp = 0.1. (a) Variation of system weight with wing area and battery-specific energy density. (b) Variation of MTOW with Wing Area and battery energy-specific density.

Figure 4-6 explores the effect of aircraft electrification on fuel burn. It has to be noted that the effect of additional drag due to aircraft electrification has not been considered in this case study. However, as the aircraft has a low cruise speed and only 19 passengers, the effect of additional drag due to the electrification of the bleed air system (using ram air instead of engine bleed) is negligible. Additionally, this study has not considered the effect of increased cooling requirements for electrification.

Figure 4-6a shows that even for a current battery E_{sp} of 272 Wh/kg, only the low-voltage MEA system architectures will not reduce the mission fuel burn compared to the conventional configuration. The best system architectures are high-voltage AEA configurations. Still, with current E_{sp}, even for the small hybridization of the aircraft (10%), the MTOW exceeds the limit of the Part 23 certification and is thus not viable for a retrofit, including a wing area increase.

Figure 4-6b explores the impact of increasing E_{sp} for a hybrid configuration with H_p = 0.1. One can see that all architectures with E_{sp} = 800 Wh/kg and E_{SP} = 1500 Wh/kg lead to configurations that meet the Part 23 MTOW limit and that lead to potential fuel burn reductions of 10% and more.

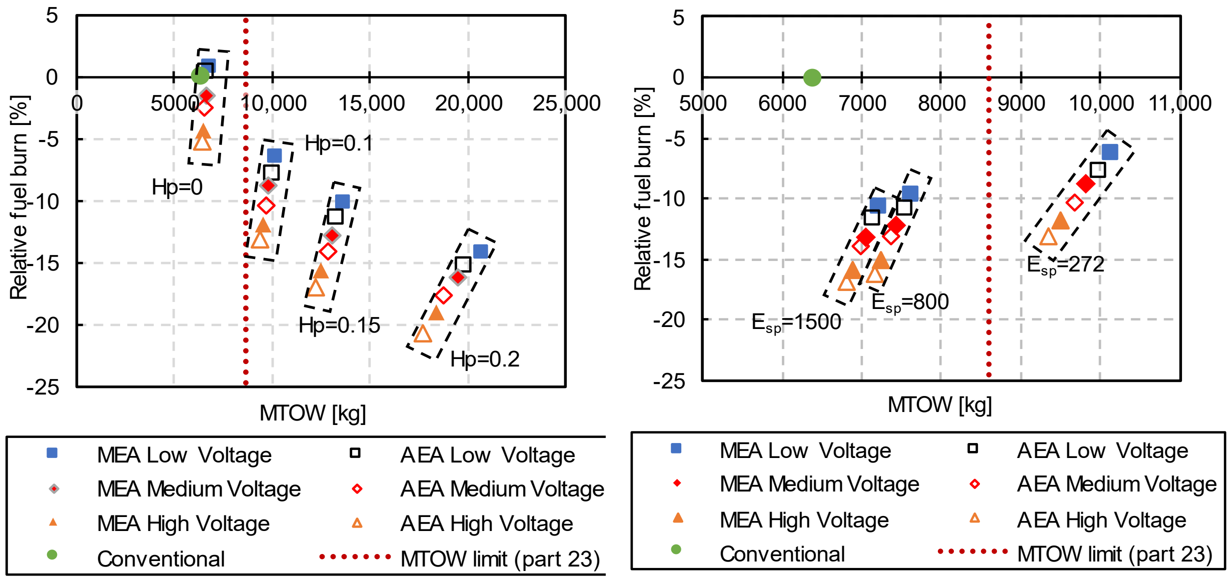


Figure 4-6 Impact analysis of the hybridization factor, battery-specific energy density, system architecture, and voltage level. (**a**) Variation of fuel burn with MTOW and hybridization factor at E_{sp} = 272 Wh/kg. (**b**) Variation of fuel burn with MTOW and battery energy-specific density at H_p = 0.1.

5 Conclusion

The aviation industry is looking at the electrification of aircraft as the next potential solution for reducing carbon emissions. The unconventional fully electric and hybrid electric aircraft configurations are being studied as the most prominent technologies capable of achieving the targets. In this context, commuter or regional aircraft are the most economically viable candidates, which are extensively studied by the industry. Hence, novel technologies in terms of architecture feasibility must be tested in novel configurations within the conceptual design phase where the cost of development can be minimal. The multidisciplinary problem can only be solved by the development of new design tools capable enough to carry out integrated analysis by incorporating all the possible novel disciplines.

This chapter summarizes the contribution of the thesis and discusses potential future work.

5.1 Contribution of thesis

The thesis has reviewed existing approaches of systems consideration in conceptual aircraft design environment. The conventional aircraft design frameworks do not integrate system aspects well enough to explore novel technologies and system architectures at the right level of detail. Recently introduced multidisciplinary analysis frameworks are well-suitable for analyzing multiple architecture configurations. However, the underlying mathematical models for the subsystems are not publicly available or have restricted access. Hence, it is required to develop a sandbox framework capable enough to analyze the integrated novel subsystems effect at the aircraft level.

The thesis presents a Multidisciplinary Analysis (MDA) framework for aircraft systems integration within the scope of aircraft conceptual design. The described generic MDA methodology is adaptable to future aircraft and systems configurations. The proposed methodology covers commuter, business and regional aircraft. The overall subsystem modelling consists of physics-based and empirical methods to study the architecture variants, such as technology changes and electrification levels at the aircraft level. Various validation cases are studied, and the validation results indicate that the Systems MDA results are well within the acceptable range of conceptual design. Furthermore, the granular system architecture definition approach introduced in this thesis enables the integration of safety assessment using the ASSESS L0 tool.

A case study on the DO-228 commuter aircraft is set up to demonstrate the capabilities of the overall framework. The case study explores the simultaneous hybridization and electrification of DO-228. Hybridization includes electrification of the propulsion system by introducing dedicated energy storage (batteries) for propulsive needs with respect to the hybridization factor and the energy density as per battery technology. To demonstrate the functionality of the framework, parametric analysis is conducted between the hybridization factor, system architecture electrification level and subsystem technologies such as flight control actuation technology, voltage level in the electrical power system and the altering battery energy densities.

The framework offers valuable insight into the connection between design decisions made at the aircraft level and cascade effects on the systems and subsystems. Additionally, the proposed framework offers a sandbox for the integration of more sophisticated systems such as safety, thermal and maintenance analyses[77].

5.2 Limitations of the framework

Although the implemented framework can conduct complex MDA analysis, it has with several limitations:-

- Although the tool can estimate subsystem effects at the aircraft level, the physics-based estimation is subject to error. The sources of error are in the individual component sizing models.
- Not all subsystems are implemented with a physics-based model. The subsystems such as Galley, Furnishings, Entertainment, Avionics, Instruments, Environmental Control System, Ice Protection System and Landing Gear System are yet to be expanded based on physics-based method.
- Subsystems are currently quantified in terms of power and weight. However, the placement of subsystems in terms of available volume is not yet integrated.
- Current wiring and hydraulic line routing are empirical. However, adaptive architecture has not yet been implemented.

5.3 Future work

Future research will concentrate on enlarging the framework to represent all essential subsystems in enough detail to capture the effects of secondary power off-take, the drag penalty due to increased ventilation flow requirements in electrified aircraft, and better capture the changes in the propulsion subsystem. The current MDA framework will be connected to the top-level aircraft analysis to carry out MDAO analysis. This will allow additional trade-off analysis capability to capture the most prominent configurations automatically.

The current safety analysis within the aircraft systems module consists of minimal safety analysis such as electrical system generation, hydraulic system and flight control system failure scenarios. However, in terms of electrification and the addition of energy storage, additional failure scenarios must be included. For this purpose, integration of safety tool ASSESS (Aircraft System Safety Assessment) [22,23] is an additional effort committed by the Lab. Although the flight control and electrical systems are subject to physics-based modelling in the systems sizing workflow, error propagation across various sources still occurs. The physics-based modelling techniques involves connections between several subsystems or components, or even mathematical models itself within the subsystem. The error propagation in the models corresponds to uncertainty towards aircraft level analysis. It is essential to integrate uncertainty analysis to interpret these outcomes especially when using novel technologies data which cannot be validated.

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List of Publications

Journal paper

Mohan, V., Jeyaraj, A. K., and Liscouët-Hanke, S., “Systems Integration Framework for Hybrid-Electric Commuter and Regional Aircraft,” *Aerospace*, Vol. 10, No. 6, 2023, p. 533. https://doi.org/10.3390/aerospace10060533

Conference paper

Mohan, V., Jeyaraj, A. K., and Liscouet-Hanke, S., “Systems Integration Considerations for Hybrid-Electric Commuter Aircraft: Case Study for the DO-228,” 2023. https://doi.org/10.2514/6.2023-1361

References

[1] Collier, F., and Wahls, R., “NASA Aeronautics Strategic Implementation Plan Strategic Thrust 3A Roadmap Overview Ultra-Efficient Commercial Vehicles-Subsonic Transports,” 2016. Retrieved 17 May 2022. https://nari.arc.nasa.gov/sites/default/files/attachments/Thrust%203a_1june2016-web.pdf

[2] Bruner, S., Baber, S., Harris, C., Caldwell, N., Keding, P., Rahrig, K., Pho, L., and Wlezian, R., “NASA N + 3 Subsonic Fixed Wing Silent Efficient Low-Emissions Commercial Transport (SELECT) Vehicle Study Revision A,” No. NASA Technical Report 2010-216798,2010.

[3] Funke, C. C., “Concurrent Engineering in the Aircraft Industry and It’s Relationship to the Development Process,” 1997. https://doi.org/10.4271/972245

[4] Ian Moir, and Allan Seabridge, “Aircraft Systems Mechanical, Electrical, and Avionics Subsystems Integration,” 2008. https://doi.org/10.1017/CBO9781107415324.004

[5] Allan, S., and Ian, M., “Design and Development of Aircraft Systems,” John Wiley & Sons Ltd., 2020, Hoboken, NJ, USA.

[6] “Federal Aviation Administration Joint Aircraft System/Component Code Table and Definitions,” Federal Aviation Administration Flight Standards Service Regulatory Support Division Aviation Data Systems Branch, Afs-620, 2008.

[7] Liscouët-Hanke, S., “A Model-Based Methodology for Integrated Preliminary Sizing and Analysis of Aircraft Power System Architectures,” Ph.D. dissertation. Institut National des Sciences Appliquée Toulouse, Toulouse, France, 2008.

[8] Moir, I., and Seabridge, A., “Aircraft Systems Mechanical, Electrical, and Avionics Subsystems Integration,” *AIAA Aerospace Sciences Meeting, 2018*, Vol. 3, No. 3, 2019, pp. 1–12. https://doi.org/10.1017/CBO9781107415324.004

[9] Dollmayer, J., and Carl, U. B., “Consideration Of Fuel Consumption Caused By Aircraft Systems in Aircraft Design,” *25th International Congress Of The Aeronautical Sciences*, 2006, pp. 1–10.

[10] Brelje, B. J., and Martins, J. R. R. A., “Electric, Hybrid, and Turboelectric Fixed-Wing Aircraft: A Review of Concepts, Models, and Design Approaches,” *Progress in Aerospace Sciences*, Vol. 104, 2019, pp. 1–19. https://doi.org/10.1016/j.paerosci.2018.06.004

[11] Zhu, Q., Forsyth, A., and Todd, R., “Investigation of Hybrid Electric Aircraft Operation on Battery Degradation,” 2019. https://doi.org/10.1109/ESARS-ITEC.2018.8607617

[12] el Kharbachi, A., Zavorotynska, O., Latroche, M., Cuevas, F., Yartys, V., and Fichtner, M., “Exploits, Advances and Challenges Benefiting beyond Li-Ion Battery Technologies,” *Journal of Alloys and Compounds*, Vol. 817, 2020, p. 153261. https://doi.org/10.1016/j.jallcom.2019.153261

[13] Ralph H, J., Cheryl. Bowman, Amy, J., Rodger, D., and James L, F., “Overview of NASA Electrified Aircraft Propulsion Research for Large Subsonic Transports,” 2017. Retrieved 26 May 2022. https://ntrs.nasa.gov/citations/20170012222

[14] Charles, A., “Faradair Establishes Duxford HQ for Hybrid Electric STOL,” ainonline.com, Jul 16 2020.

[15] Bill Howard, “The Chevy Volt of Airplanes, the Volta Volare GT4, Is Ready to Fly,” ExtremeTech, 2012.

[16] Air Canada, “Air Canada to Acquire 30 ES-30 Electric Regional Aircraft from Heart Aerospace,” Montreal, Sep2022. Retrieved 1 November 2023. https://www.newswire.ca/news-releases/air-canada-to-acquire-30-es-30-electric-regional-aircraft-from-heart-aerospace-873505575.html

[17] Matt Knapp, and James Provost, “Zunum Aeros’s Hybrid Electric Airplane Aims to Rejuvenate Regional Travel,” IEEE Spectrum, Kirkland, Washington, Apr 26 2018.

[18] Chin, J., Aretskin-Hariton, E., Ingraham, D., Hall, D., Schnulo, S. L., Gray, J., and Hendricks, E. S., “Battery Evaluation Profiles for X-57 and Future Urban Electric Aircraft,” 2020. https://doi.org/10.2514/6.2020-3567

[19] Boulanger, V., Gaillardetz, A., Chiasson Poirier, L., Guimond, K., Lachance, F., Raby, E., and Rancourt, D., “Lessons Learned from the Design, Manufacturing, and Test of an All-Electric General Aviation Aircraft,” 2020. https://doi.org/10.2514/6.2020-2652

[20] Chen, Z., and Fielding, J. P., “A Zonal Safety Analysis Methodology for Preliminary Aircraft Systems and Structural Design,” *The Aeronautical Journal*, Vol. 122, No. 1255, 2018, pp. 1330–1351. https://doi.org/10.1017/aer.2018.58

[21] Davies, M., “The Standard Handbook for Aeronautical and Astronautical Engineers,” McGraw-Hill Education, 2002.

[22] Jeyaraj, A. K., and Liscouët-Hanke, S., “A Safety-Focused System Architecting Framework for the Conceptual Design of Aircraft Systems,” *Aerospace*, No. 9, 2022. https://doi.org/10.3390/aerospace9120791

[23] Jeyaraj, A. K., Bussemaker, J. H., Liscouët-Hanke, S., and Boggero, L., “Systems Architecting: A Practical Example of Design Space Modeling and Safety-Based Filtering within the Agile 4.0 Project,” *33rd Congress of the International Council of Aeronautical Science*, No. Stockholm, Sweden, 2022. https://www.agile-project.eu/

[24] Sanchez, F., Liscouët-Hanke, S., and Tfaily, A., “Improving Aircraft Conceptual Design through Parametric CAD Modellers – A Case Study for Thermal Analysis of Aircraft Systems,” *Computers in Industry*, Vol. 130, 2021. https://doi.org/10.1016/j.compind.2021.103467

[25] Jan, R., “Airplane Design,” DARcorporation, Lawrence, Kansas, 1985.

[26] Torenbeek, E., “Synthesis of Subsonic Airplane Design,” Delft University Press, Delft, The Netherlands, 1982.

[27] Raymer, D., “Aircraft Design: A Conceptual Approach,” AIAA Education Series, Washington. DC, 1989.

[28] Howe, D., “Aircraft Conceptual Design Synthesis,” Professional Engineering Publishing Limited, London, UK, 2000.

[29] Raymer, D., “Aircraft Design: A Conceptual Approach, Sixth Edition,” *Aircraft Design: A Conceptual Approach, Sixth Edition*, 2018. https://doi.org/10.2514/4.104909

[30] Wells, D. P., and Linwood A. McCullers, “The Flight Optimization System Weights Estimation Method,” *NASA/TM/2017-179627*, Vol. I, No. June, 2017.

[31] Horvath, B. L., and Wells, D. P., “Comparison of Aircraft Conceptual Design Weight Estimation Methods to the Flight Optimization System,” *AIAA Aerospace Sciences Meeting, 2018*, No. 210059, 2018. https://doi.org/10.2514/6.2018-2032

[32] Koeppen, C., and Carl, U. B., “Kopplungen Und Sensitivitäten Des Einflusses Von Systemen Auf Den Flugzeugentwurf,” Vol. 11, 2003, pp. 17–20. https://doi.org/10.15480/882.218

[33] Lammering, T., “Integration of Aircraft Systems into Conceptual Design Synthesis,” Ph.D. dissertation. Rhenish-Westphalian Technical University of Aachen, North Rhine-Westphalia, Germany, 2014.

[34] Chakraborty, I., “Subsystem Architecture Sizing and Analysis for Aircraft Conceptual Design,” Ph.D. dissertation. Georgia Institute of Technology, Atlanta, Georgia, 2015.

[35] Daniel, P. R., “Enhancing Aircraft Conceptual Design Using Multidisciplinary Optimization,” Ph.D. dissertation. Swedish Royal Institute of Technology, Sweden, 2002.

[36] Werner-Westphal, C., Heinze, W., and Horst, P., “Multidisciplinary Integrated Preliminary Design Applied to Unconventional Aircraft Configurations,” *Journal of Aircraft*, Vol. 45, No. 2, 2008, pp. 581–590. https://doi.org/10.2514/1.32138

[37] Heath, C., and Gray, J., “OpenMDAO: Framework for Flexible Multidisciplinary Design, Analysis and Optimization Methods,” 2012. https://doi.org/10.2514/6.2012-1673

[38] Moore, K., Naylor, B., and Gray, J., “The Development of an Open Source Framework for Multidisciplinary Analysis and Optimization,” 2008. https://doi.org/10.2514/6.2008-6069

[39] Gray, J. S., Hwang, J. T., Martins, J. R. R. A., Moore, K. T., and Naylor, B. A., “OpenMDAO: An Open-Source Framework for Multidisciplinary Design, Analysis, and Optimization,” *Structural and Multidisciplinary Optimization*, Vol. 59, No. 4, 2019, pp. 1075–1104. https://doi.org/10.1007/s00158-019-02211-z

[40] Chiesa, S., Fioriti, M., and Viol, N., “Methodology for an Integrated Definition of a System and Its Subsystems: The Case-Study of an Airplane and Its Subsystems",” *Systems Engineering - Practice and Theory*, 2012. https://doi.org/10.5772/34453

[41] Boggero, L., Fioriti, M., Corpino, S., and Ciampa, P. D., “On-Board Systems Preliminary Sizing in an Overall Aircraft Design Environment,” *17th AIAA Aviation Technology, Integration, and Operations Conference,* No. Denver, Colorado, 2017, pp. 1–13. https://doi.org/10.2514/6.2017-3065

[42] van Gent, I., Aigner, B., Beijer, B., Jepsen, J., and la Rocca, G., “Knowledge Architecture Supporting the next Generation of MDO in the AGILE Paradigm,” *Progress in Aerospace Sciences*, Vol. 119, 2020, p. 100642. https://doi.org/10.1016/j.paerosci.2020.100642

[43] Fioriti, M., Della Vecchia, P., Donelli, G., and Hansmann, P., “Assessing the Integration of Electrified On-Board Systems in an MDAO Framework for a Small Transport Aircraft,” *AIAA Aviation Forum, 2021, Dalla, Texas*. https://doi.org/10.2514/6.2021-3094

[44] Jünemann, M., Thielecke, F., Peter, F., Hornung, M., Schültke, F., and Stumpf, E., "Methodology for Design and Evaluation of More Electric Aircraft Systems Architectures within the AVACON Project," *Deutscher Luft- und Raumfahrtkongress*, 2019, pp. 1–18. https://doi.org/10.25967/480197

[45] Raymer, D. P., "Vehicle Scaling Laws for Multidisciplinary Optimization (Preliminary Report)," *39th Aerospace Sciences Meeting and Exhibit*, No. October, 2001. https://doi.org/10.2514/6.2001-532

[46] Liscouët-Hanke, S., and Huynh, K., "A Methodology for Systems Integration in Aircraft Conceptual Design - Estimation of Required Space," *SAE Technical Papers*, Vol. 7, 2013. https://doi.org/10.4271/2013-01-2235

[47] Budinger, M., "Preliminary Design and Sizing of Actuation Systems," *Mechanical engineering [physics.class-ph].UPSToulouse,2014. tel-01112448*.

[48] Tfaily, A., Liscouët-Hanke, S., and Esdras, G., "Parametric 3D Modeling for Integration of Aircraft Systems in Conceptual Design," *Canadian Aeronautics and Space Institute Conference*, 2015.

[49] Lambe, A. B., and Martins, J. R. R. A., "Extensions to the Design Structure Matrix for the Description of Multidisciplinary Design, Analysis, and Optimization Processes," *Structural and Multidisciplinary Optimization*, Vol. 46, No. 2, 2012, pp. 273–284. https://doi.org/10.1007/s00158-012-0763-y

[50] Mohan, V., Jeyaraj, A. K., and Liscouët-Hanke, S., "Systems Integration Framework for Hybrid-Electric Commuter and Regional Aircraft," *Aerospace*, Vol. 10, No. 6, 2023, p. 533. https://doi.org/10.3390/aerospace10060533

[51] Mohan, V., Jeyaraj, A. K., and Liscouet-Hanke, S., "Systems Integration Considerations for Hybrid-Electric Commuter Aircraft: Case Study for the DO-228," *AIAA Scitech Forum, 2023, National Harbor, MD*. https://doi.org/10.2514/6.2023-1361

[52] Boden, B., Flink, J., Först, N., Mischke, R., Schaffert, K., Weinert, A., Wohlan, A., and Schreiber, A., "RCE: An Integration Environment for Engineering and Science," *SoftwareX*, Vol. 15, 2021, p. 100759. https://doi.org/10.1016/j.softx.2021.100759

[53] Licheva, G., and Liscouët-Hanke, S., "A Conceptual Sizing Tool for Regional and Commuter Aircraft with Hybrid-Electric Propulsion ," *AIAA Scitech Forum 2023, National Harbor, MD*. https://doi.org/10.2514/6.2023-0213

[54] Heit, Z., and Liscouët-Hanke, S., "Estimation of Battery Pack Layout and Dimensions for the Conceptual Design of Hybrid-Electric Aircraft," *AIAA Scitech Forum 2023, National Harbor, MD*. https://doi.org/10.2514/6.2023-1362

[55] Rodriguez, C. D., and Liscouët-Hanke, S., "Architecture-Based Fuel System Conceptual Design Tool for Hybrid-Electric Aircraft," *AIAA Aviation Forum, 2021, Virtual Event*. https://doi.org/10.2514/6.2021-2408

[56] William, J. E., and Vukelich, S. R., "The USAF Stability and Control Digital DATCOM Volume I, Users Manual," Vol. I, No. Technical Report AFDL-TR-79-3032, 1979. Retrieved 28 May 2022. https://ntrl.ntis.gov/NTRL/dashboard/searchResults/titleDetail/ADA086557.xhtml

[57] Anderson, R. D., Flora, C. C., Nelson, R. M., E.T.Raymond, and J.H.Vincent, “Development of Weight and Cost Estimates for Lifting Surfaces with Active Controls,” *NASA Technical Report N76-20150*, 1979.

[58] SAE International, “Description of Actuation Systems for Aircraft With Fly-By-Wire Flight Control Systems,” *AIR4253B, SAE International: Warrendale, PA, USA*, 2018. https://doi.org/https://doi.org/10.4271/AIR4253B

[59] SAE International, “Mechanical Control Design Guide,” 2012.

[60] Maré, J.-C., “Aerospace Actuators 3,” Wiley-ISTE Ltd, UK, 2018. https://doi.org/10.1002/9781119505433

[61] Obert, E., “Aerodynamic Design of Transport Aircraft,” IOS Press, Amsterdam, 2009.

[62] SAE International, “Commercial Aircraft Hydraulic System Sizing,” *ARP6277, SAE International: Warrendale, PA, USA*, 2021. https://doi.org/https://doi.org/10.4271/ARP6277

[63] SAE International, “Fundamentals in Wiring Selection and Sizing for Aerospace Application,” *AIR6540 International: Warrendale, PA, USA*, 2019. https://doi.org/https://doi.org/10.4271/AIR6540

[64] SAE International, “Wiring Aerospace Vehicle,” *AS50881, SAE International: Warrendale, PA, USA*, 1998. https://doi.org/10.4271/AS50881

[65] Brombach, J., Schröter, T., Lücken, A., and Schulz, D., “Optimizing the Weight of an Aircraft Power Supply System through a +/- 270 Vdc Main Voltage,” *Przeglad Elektrotechniczny*, Vol. 88, No. 1 A, 2012, pp. 47–50.

[66] Brombach, J., Schröter, T., Lücken, A., and Schulz, D., “Optimized Cabin Power Supply with a / 270 v DC Grid on a Modern Aircraft,” *2011 7th International Conference-Workshop Compatibility and Power Electronics, CPE 2011* , No. Tallinn, Estonia, 2011, pp. 425–428. https://doi.org/10.1109/CPE.2011.5942274

[67] Esdras, G. F., and Liscouët-Hanke, S., “An Electrical Load Estimation Tool for Aircraft Conceptual Design,” *SAE Technical Papers*, Vol. 7, 2013. https://doi.org/10.4271/2013-01-2206

[68] Bagshaw, M., and Illig, P., “The Aircraft Cabin Environment,” *Travel Medicine*, Elsevier, 2019, pp. 429–436. https://doi.org/10.1016/B978-0-323-54696-6.00047-1

[69] SAE International, “Aircraft Fuel Weight Penalty Due to Air Conditioning,” *AIR1168/8A, SAE International: Warrendale, PA, USA*, 2011. https://doi.org/https://doi.org/10.4271/AIR1168/8A

[70] Scholz, D., Seresinhe, R., Staack, I., and Lawson, C., “Fuel Consumption Due To Shaft Power Off-Takes From the Engine,” *4th International Workshop on Aircraft System Technologies (AST 2013), 23-24 April 2013, Hamburg, Germany*, 2013, pp. 169–179.

[71] Alder, M., Moerland, E., Jepsen, J., and Nagel, B., “Recent Advances in Establishing a Common Language for Aircraft Design with CPACS,” *In Proceedings of the Aerospace Europe Conference, Bordeaux, Frankreich, 25–28 February*, 2020.

[72] Taylor, J. W. R., “Jane’s All the World’s Aircraft 1982-1983,” Janes Information Services: Croydon, UK, 1982.

[73] Hindustan Aeronautics Limited, “Technical Specification for HAL-Dornier-228-202 Aircraft System,” *Transport Aircraft Division, Banglore, India*, 2014. Retrieved 28 May 2022. https://www.hal-india.co.in/Common/Uploads/TenderDoc/26353_TenderPDF2_Annexure-1.pdf

[74] Finger, D. F., de Vries, R., Vos, R., Braun, C., and Bil, C., “Cross-Validation of Hybrid-Electric Aircraft Sizing Methods,” *Journal of Aircraft*, 2022, pp. 1–19. https://doi.org/10.2514/1.c035907

[75] Vladimirov, S. V., and Forde, S., “Demonstration Program to Design, Manufacture and Test an Autonomous Electro-Hydrostatic Actuator to Gimbal Large Booster-Class Engines,” *Collection of Technical Papers - AIAA/ASME/SAE/ASEE 42nd Joint Propulsion Conference*, Vol. 1, No. July 2006, 2006, pp. 541–546. https://doi.org/10.2514/6.2006-4364

[76] Eftekhari, A., “On the Theoretical Capacity/Energy of Lithium Batteries and Their Counterparts,” *ACS Sustainable Chemistry & Engineering*, Vol. 7, No. 4, 2019, pp. 3684–3687. https://doi.org/10.1021/acssuschemeng.7b04330

[77] Postnikov, S. E., Trofimov, A. A., and Baikov, S. V., “Architecture Options Estimate for the Near-Medium-Haul Aircraft Control System by the Reliability, Mass and Power Consumption Criteria,” *Aerospace Systems*, Vol. 2, No. 1, 2019, pp. 33–40. https://doi.org/10.1007/s42401-018-0017-9

[78] Eaton, “Eaton ’s Aerospace Product Capabilities in A320/321,” Eaton Aerospace Group, Apr2014. Retrieved 20 November 2022. https://www.eaton.com/content/dam/eaton/markets/aerospace/Airbus-A318-A319-A320-A321-capabilities-brochure.pdf

[79] Cremer, R., and Engler, A., “Optimizing Power Electronics for the More Electric Aircraft,” *SAE Technical Paper Series, SAE International*, 2014, p. https://doi.org/10.4271/2014-01-2194

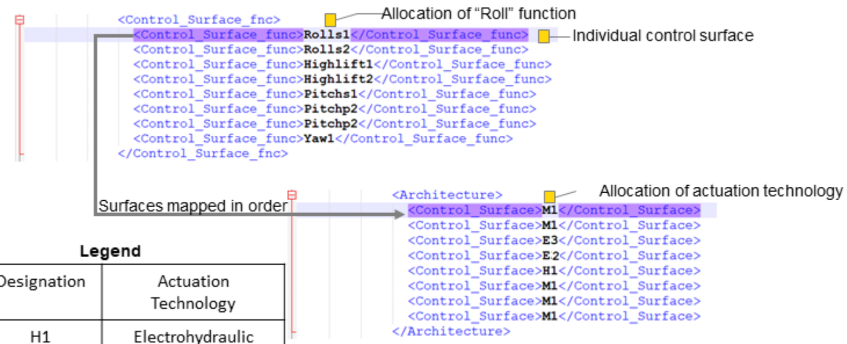
[80] Qiao, G., Liu, G., Shi, Z., Wang, Y., Ma, S., and Lim, T. C., “A Review of Electromechanical Actuators for More/All Electric Aircraft Systems,” *Proceedings of the Institution of Mechanical Engineers, Part C: Journal of Mechanical Engineering Science*, Vol. 232, No. 22, 2018, pp. 4128–4151. https://doi.org/10.1177/0954406217749869

[81] Zaccai, D., Bertels, F., and Vos, R., “Design Methodology for Trailing-Edge High-Lift Mechanisms,” *CEAS Aeronautical Journal*, Vol. 7, No. 4, 2016, pp. 521–534. https://doi.org/10.1007/s13272-016-0202-7

[82] Madonna, V., Giangrande, P., and Galea, M., “Electrical Power Generation in Aircraft: Review, Challenges, and Opportunities,” *IEEE Transactions on Transportation Electrification*, Vol. 4, No. 3, 2018, pp. 646–659. https://doi.org/10.1109/TTE.2018.2834142

Appendices

Appendix A Systems MDA architecture descriptor



Legend

Designation	Actuation Technology
H1	Electrohydraulic Servo Actuator
H2	Hydraulic Power Drive Unit
H3	Horizontal Stabilizer Actuator
E1	Electro-hydrostatic Actuator
E2	Electro-Mechanical Actuator
E3	Electric Flap Drive
E4	Electric Backup Hydraulic Actuator
M1	Mechanical Linkage

Figure A-1 Definition of control surfaces and assignment of functions and actuators for a FCS using the System MDA descriptor

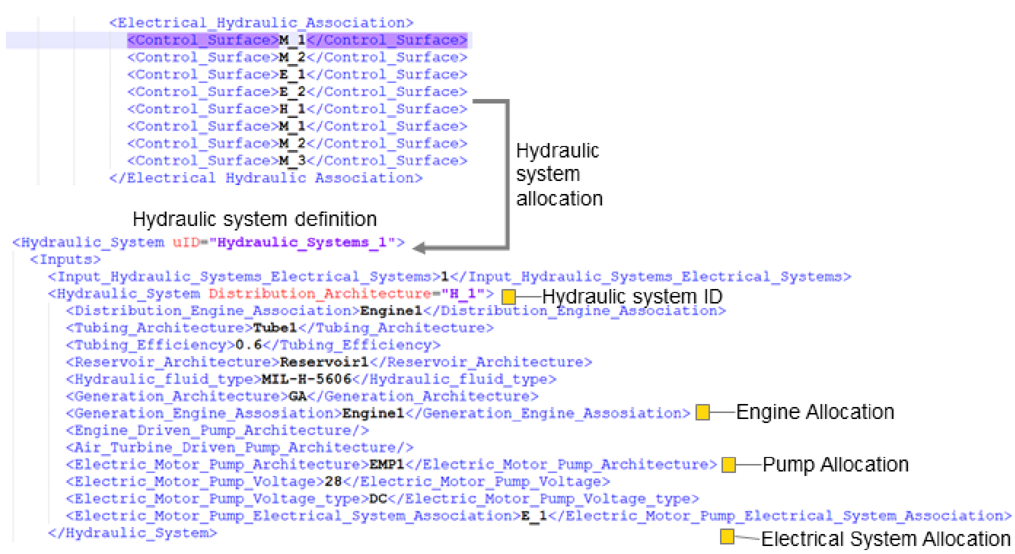


Figure A-2 Allocation and definition of a hydraulic power system to a specific control surface actuator.

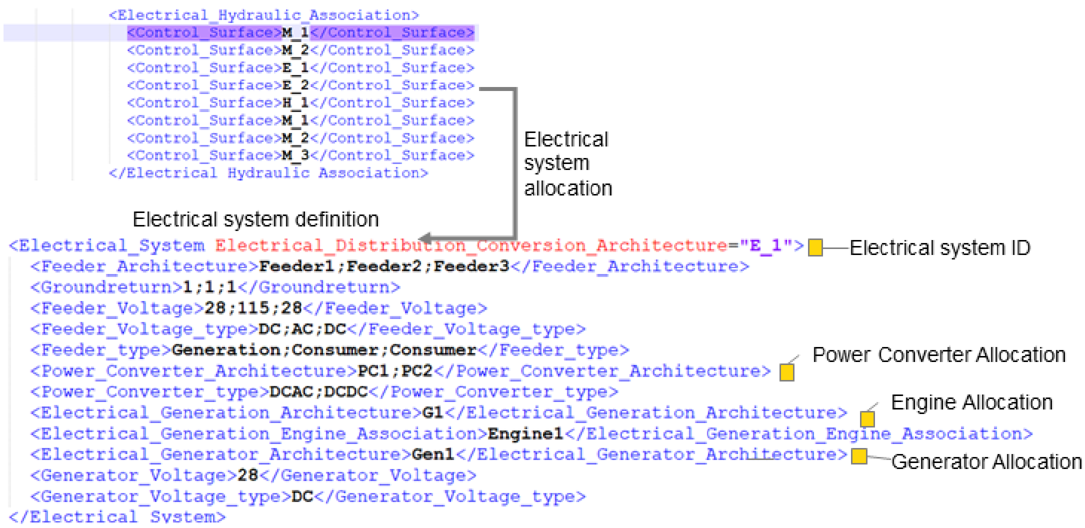



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Figure A-3 Allocation and definition of an electrical power system to a specific control surface actuator.

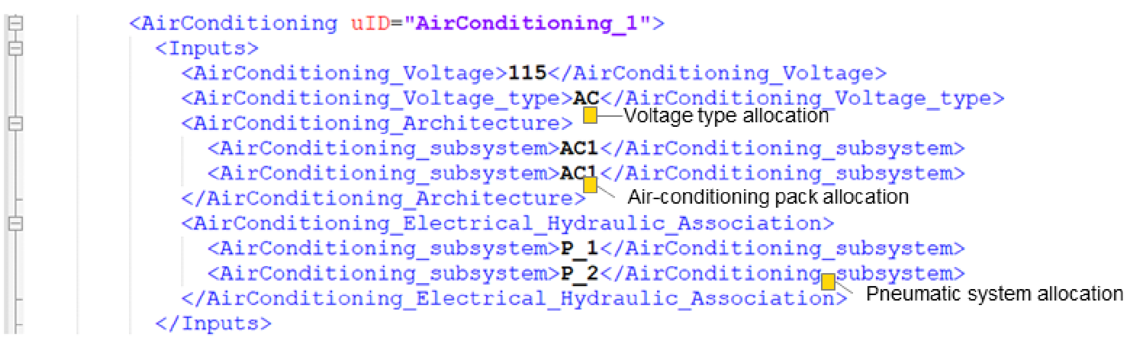


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Figure A-4 Definition of air conditioning packs within an environmental control system architecture.

Appendix B Subsystem descriptions

Table 5-1 Electric power consumer demand

Subsystem	Power Demand	
Lights	$P_{Lig,norm} = 0.31\ L_{fuselage}$	(5-1)
Avionics and Instruments	$P_{Av,inst_{taxi}} = 0.612\ e^{0.048\ L_{fuselage}}$	(5-2)
	$P_{Av,inst_{nom}} = 0.02\ L_{fuselage}{}^{1.55}$	(5-3)
Ice Protection[1]	$P_{ips,nom} = 0.035\ s_w + 2.02$	(5-4)
Air-conditioning[2] [67]	$P_{ac,nom} = 0.077\ V_{cabin} - 0.40$	(5-5)
Fuel System[3]	$P_{FS,nom} = 2.88\ e^{\frac{0.0399 L_{fuselage}}{N_{eng}}}$	(5-6)

Table 5-2 Control surface deflection

Control Surface Type	Control Surface Function	Surface Deflection Rate (deg/s)
Aileron, Spoiler	Roll	60
Spoiler	Autobrake	30
Flaps, Slats	High-lift	10
Rudder	Yaw	40
Elevator	Pitch	60
Horizontal Stabilizer	Pitch trim	0.5

Table 5-3 Control surface usage factor

Control Surface Type	Control Surface Function	Ground	Taxi	Takeoff	Climb	Cruise	Descent	Approach	Landing
Aileron, Spoiler	Roll	0.33	0.33	0	1	1	1	0.33	0.33

[1] This does not include electric de-icing system. Electric de-icing power demand can be considered to be 5% $P_{ips,nom}$.
[2] This power demand corresponds to conventional aircraft air-conditioning units electrical power demand.
[3] The *Fuel System* power demands are estimated using this equation whereas the *Fuel System* weights are obtained using the physics-based methods described in [55]

Spoiler	Autobrake	0.33	0.33	0	1	1	1	0	0.33
Flaps, Slats	High-lift	0	1	0	0	0	1	0	0
Rudder	Yaw	0.33	0.33	0.33	0.33	0.33	0.5	0.5	0.33
Elevator, Horizontal Stabilizer	Pitch	0	0	1	1	0.15	1	0.33	1

Table 5-4 Subsystem P/W ratio

Subsystem	P/W Ratio	Ref.	Subsystem	P/W Ratio	Ref.
HMA	0.4 $\mathrm{kW/kg}$	[77]	EDP	4.6 $\mathrm{kW/kg}$	[60]
EHSA	0.2 $\mathrm{kW/kg}$	[77]	HMG	0.46 $\mathrm{kW/kg}$	[78]
EHA	0.2 $\mathrm{kW/kg}$	[77]	Actuator Power Electronics	2 $\mathrm{kW/kg}$	[79]
EMA	0.40 $\mathrm{kW/kg}$	[80]	EMP	0.4 $\mathrm{kW/kg}$	[60]
Hydraulic Power Drive	0.043 $\mathrm{kW/kg}$	[81]	AC Generator	1.3 $\mathrm{kVA/kg}$	[82]
Electric Motor Power Drive	0.045 $\mathrm{kW/kg}$	[81]	DC Generator	0.5 $\mathrm{kW/kg}$	[82]
THSA	0.045 $\mathrm{kW/kg}$	[60]	EBHA	0.13 $\mathrm{kW/kg}$	[75]

Table 5-5 Aircraft parameters

Aircraft Parameter	Value	Aircraft Parameter	Value
Wing span $[\mathrm{m}]$	16.97	Number of passengers	19
Payload [kg]	1959.9	Number of crew	2
Range $[\mathrm{m}]$	395,935.2	Hybrid propulsion architecture	Parallel
Mach	0.4	Service ceiling [ft]	25,000
Aspect ratio	9.0	Hydraulic system pressure $[\mathrm{psi}]$	3000
Length $[\mathrm{m}]$	16.56	Generation voltage $[\mathrm{V}]$	230
Cabin volume $[\mathrm{m}^3]$	14.7	Generation voltage type	AC
Number of engines	2	Inverter voltage $[\mathrm{V}]$	115
Propeller diameter $[\mathrm{m}]$	2.73	Wing area $[\mathrm{m}^2]$	32

Fuselage length [m]	16.56	Fuselage width [m]	1.346
Vertical stab length [m]	2.63	Horizontal tail length [m]	6.45
Engine configuration	Wing mounted	Diameter of engine [m]	0.98
Length of engine [m]	2.76		

Table 5-6 Conventional vs. hybridized aircraft systems architecture

Parameter	**Conventional**	**Hp = 0.1**
FCS primary flight control actuator	Mechanical	EHA, EHSA
FCS secondary flight control actuator	Mechanical	All EMA
Power converter	DCAC, DCDC	ACDC, ACAC
FCS voltage (V)	28	28
FCS voltage type	DC	DC
Avionics voltage (V)	28	28
FCS voltage type	DC	DC
Other systems voltage (V)	115	230
Other systems voltage type	AC	AC
Electrical generation voltage (V)	28	230
Electrical generation voltage type	DC	AC

Table 5-7 Subsystem power demand ratios

Flight Phase / **Subsystem**	**Ground**	**Taxi**	**Takeoff**	**Climb**	**Cruise**	**Descent**	**Approach**	**Landing**
Galley, Entertainment, Furnishing	0.46	1.03	1	1	1.09	0.72	0.72	0.72
Lights	0.945	0.75	1	1	0.943	1.19	1.19	1.07
Avionics, Instruments	0.25[4]	1[5]	1	1	1	1	1	1
Ice Protection	0.33	0.33	0.33	0.33	0.33	0.5	0.5	0.33
Air-conditioning	1	1	1.06	1	1	1.06	1.06	0.92
Fuel System	0	0	1	1	0.15	1	0.33	1

[4] Unless specified P_{norm} is to be used with factors to estimate power demand per flight phase
[5] Factor to be used with equation (3-19)

Appendix C Systems identifiers

Table 5-8 Aircraft Systems classification based on ATA [6]

ATA Chapter	Aircraft Systems
21	Air Conditioning & Pressurization
22	Auto flight
23	Communication
24	Electrical Power
25	Equipment and Furnishing
26	Fire Protection
27	Flight Controls
28	Fuel
29	Hydraulic Power
30	Ice and Rain Protection
31	Instrumentation, Indication & Recording
32	Landing Gear
33	Lights
34	Navigation
35	Oxygen
36	Pneumatic
38	Water & Waste
39	Electrical, Electronic Panels & Multipurpose Components
42	Integrated Modular Avionics
44	Cabin Systems
49	Auxiliary Power System