Design techniques to support aircraft systems development in a collaborative MDO environment

Doctoral Dissertation

Doctoral Program in Aerospace Engineering (30th Cycle)

Design techniques to support aircraft systems development in a collaborative MDO environment

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Politecnico di Torino
2018

Declaration

I hereby declare that, the contents and organization of this dissertation constitute my own original work and does not compromise in any way the rights of third parties, including those relating to the security of personal data.

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2018

* This dissertation is presented in partial fulfillment of the requirements for **Ph.D. degree** in the Graduate School of Politecnico di Torino (ScuDo)

Acknowledgment

I have carried out the doctorate within the context of the Italian national project TIVANO, which has been funded by the Italian Ministry of Education, University and Research. Thanks to this project, I had the opportunity of fulfilling my research activities at Politecnico di Torino, at Leonardo – Aircraft Division in Turin and at the German Aerospace Center DLR in Hamburg. In each place I met very special people, whom I would like to acknowledge.

In particular, I thank my thesis supervisors Prof. Sabrina Corpino and Prof. Nicole Viola for their support, suggestions and contributes to my research. Many thanks to all the office mates, especially Marco, Francesca, Sara, Roberta, Davide, Valeria and Fabrizio, for the good times I spent with them in these years. Furthermore, I would like to acknowledge all the professors, researchers and PhD students who have been involved within the TIVANO project and with whom I had meetings, research activities and collaborations.

A special thanks goes to Elena Valfrè and Marco Baroero of Leonardo – Aircraft Division, for giving me the opportunity of working within an industrial context and for their advice. During my training in Leonardo I have also been advised by Marco Foglino, Bruno Di Giandomenico, Guido Pavan and Luciana Loverde. To them goes my gratitude for having shared with me their time, knowledge and experience. I am thankful also to all the people I met and joined in Leonardo, especially all the office colleagues.

I would like to say thanks – or even better, "*danke schön*" – to all the researchers I met during the three-month internship I spent at the Integrated Aircraft Design department of the DLR of Hamburg. Special acknowledgments go to Pier Davide Ciampa and Björn Nagel for having hosted me and to Matthias and Prajwal for having shared with me their office (and the Indian food). Thanks to all of them for their tips concerning my research activity, but also about the many attractions to visit in Hamburg.

I owe my gratitude to my parents, Piera and Stefano, for having encouraged me and supported every decision and choice I have taken all along my studies and my doctorate. I would like to thank them and dedicate my thesis.

Thanks to everyone who will be willing to read my dissertation, wishing to receive many feedbacks and comments, and hoping to having contributed as much as possible to the aircraft design discipline.

Abstract

The aircraft design is a complex multidisciplinary and collaborative process. Thousands of disciplinary experts with different design competences are involved within the whole development process. The design disciplines are often in contrast with each other, as their objectives might be not coincident, entailing compromises for the determination of the global optimal solution. Therefore, Multidisciplinary Design and Optimization (MDO) algorithms are being developed to mathematically overcome the divergences among the design disciplines. However, a MDO formulation might identify an optimal solution, but it could be not sufficient to ensure the success of a project. The success of a new project depends on two factors. The first one is relative to the aeronautical product, which has to be compliant with all the capabilities actually demanded by the stakeholders. Furthermore, a "better" airplane may be developed in accordance with customer expectations concerning better performance, lower operating costs and fewer emissions. The second important factor refers to the competitiveness among the new designed product and all the other competitors. The Time-To-Market should be reduced to introduce in the market an innovative product earlier than the other aeronautical industries. Furthermore, development costs should be decreased to maximize profits or to sell the product at a lower price. Finally, the development process must reduce all the risks due to wrong design choices.

These two main motivations entail two main objectives of the current dissertation. The first main objective regards the assessment and development of design techniques for the integration of the aircraft subsystems conceptual design discipline within a collaborative and multidisciplinary development methodology. This methodology shall meet all the necessities required to design an optimal and competitive product. The second goal is relative to the employment of the proposed design methodology for the initial development of innovative solutions. As the design process is multidisciplinary, this thesis is focused on the on-board systems discipline, without neglecting the interactions among this discipline with all the other design disciplines. Thus, two kinds of subsystems are treated in the current

dissertation. The former deals with hybrid-electric propulsion systems installed aboard Remotely Piloted Aerial Systems (RPASs) and general aviation airplanes. The second case study is centered on More and All Electric on-board system architectures, which are characterized by the removal of the hydraulic and/or pneumatic power generation systems in favor of an enhancement of the electrical system.

The proposed design methodology is based on a Systems Engineering approach, according to which all the customer needs and required system functionalities are defined since the earliest phase of the design. The methodology is a five-step process in which several techniques are implemented for the development of a successful product. In Step 1, the design case and the requirements are defined. A Model Based Systems Engineering (MBSE) approach is adopted for the derivation and development of all the functionalities effectively required by all the involved stakeholders. All the design disciplines required in the MDO problem are then collected in Step 2. In particular, all the relations among these disciplines – in terms of inputs/outputs – are outlined, in order to facilitate their connection and the setup of the design workflow. As the present thesis is mainly focused on the on-board system design discipline, several algorithms for the preliminary sizing of conventional and innovative subsystems (included the hybrid propulsion system) are presented. In the third step, an MDO problem is outlined, determining objectives, constraints and design variables. Some design problems are analyzed in the present thesis: un-converged and converged Multidisciplinary Design Analysis (MDA), Design Of Experiments (DOE), optimization. In this regard, a new multi-objective optimization method based on the Fuzzy Logic has been developed during the doctoral research. This proposed process would define the "best" aircraft solution negotiating and relaxing some constraints and requirements characterized by a little worth from the user perspective. In Step 4, the formulation of the MDO problem is then transposed into a MDO framework. Two kinds of design frameworks are here considered. The first one is centered on the subsystems design, with the aim of preliminarily highlighting the impacts of this discipline on the entire Overall Aircraft Design (OAD) process and vice-versa. The second framework is distributed, as many disciplinary experts are involved within the design process. In this case, the level of fidelity of the several disciplinary modules is higher than the first framework, but the effort needed to setup the entire workflow is much higher. The proposed methodology ends with the investigation of the design space through the implemented framework, eventually selecting the solution of the design problem (Step 5).

The capability of the proposed methodology and design techniques is demonstrated by means of four application cases. The first case study refers to the initial definition of the physical architecture of a hybrid propulsion system based on a set of needs and capabilities demanded by the customer. The second application study is focused on the preliminary sizing of a hybrid-electric propulsion system to be installed on a retrofit version of a well-known general aviation aircraft. In the third case study, the two kinds of MDO framework previously introduced are employed to design conventional, More Electric and All Electric subsystem architectures for a 90-passenger regional jet. The last case study aims at minimizing the aircraft development costs. A Design-To-Cost approach is adopted for the design of a hybrid propulsion system.

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List of Acronyms

AACE	Association for Advancement of Cost Engineering
ACM	Air Cycle Machine
AEA	All Electric Aircraft
AGB	Accessory Gear Box
APU	Auxiliary Power Unit
AR	Aspect Ratio
ASDR	Accelerate-Stop Distance Required
ATA	Air Transport Association
BDD	Block Definition Diagram
BFL	Balanced Field Length
BPR	By-Pass Ratio
BWB	Blended Wing Body
CAU	Cold Air Unit
CCS	Commercial Cabin System
CER	Cost Estimation Relationship
CFD	Computational Fluid Dynamics
CFP	COSMIC Function Point
CIAM	Central Institute of Aviation Motor
CIPS	Cowl Ice Protection System
CoG	Center Of Gravity

CPACS	Common Parametric Aircraft Configuration Schema
CS	Certification Specification
DOC	Direct Operating Cost
DOE	Design Of Experiments
DSM	Design Structure Matrix
DTC	Design-To-Cost
EBHA	Electrical Backup Hydraulic Actuator
ECS	Environmental Control System
EDP	Engine Driven Pump
EHA	Electro-Hydrostatic Actuator
EM	Electric Motor
EMA	Electro-Mechanical Actuator
EMP	Electric Motor Pump
EPDS	Electric Power Distribution System
EPGDS	Electric Power Generation and Distribution System
FAA	Federal Aviation Administration
FCS	Flight Control System
FEM	Finite Element Method
FOD	Foreign Object Debris
GCS	Ground Control Station
HALE	High Altitude Long Endurance
HEPS	Hybrid-Electric Propulsion System

HEV	Hybrid Electric Vehicle
HPGDS	Hydraulic Power Generation and Distribution System
IBD	Internal Block Diagram
ICE	Internal Combustion Engine
IDEA	Integrated Digital Electric Aircraft
IDG	Integrated Drive Generator
IFE	In-Flight Entertainment
IM	Induction Motor
INCOSE	International Council on Systems Engineering
IPS	Ice Protection System
ISR	Intelligence, Surveillance, Reconnaissance
IT	Information Technology
JPL	Jet Propulsion Laboratory
KBE	Knowledge Based Engineering
LCC	Life Cycle Cost
LFL	Landing Field Length
M&S	Model & Simulation
MALE	Medium Altitude Long Endurance
MBSE	Model Based Systems Engineering
MDA	Multidisciplinary Design Analysis
MDAO	Multidisciplinary Design Analysis and Optimization
MDO	Multidisciplinary Design and Optimization

MEA	More Electric Aircraft
MF	Membership Function
MICADO	Multidisciplinary Integrated Conceptual Aircraft Design
MLM	Maximum Landing Mass
MLW	Maximum Landing Weight
MPGDS	Mechanical Power Generation and Distribution System
MTOM	Maximum Take-Off Mass
MTOW	Maximum Take-Off Weight
OAD	Overall Aircraft Design
OEI	One Engine Inoperative
OEM	Operating Empty Mass
OEW	Operating Empty Weight
OOSEM	Object–Oriented Systems Engineering Methodology
PBS	Product Breakdown Structure
PIDO	Process Integration and Design Optimization
PM	Permanent Magnet
PPGDS	Pneumatic Power Generation and Distribution System
PrADO	Preliminary Aircraft Design and Optimization
RAMS	Reliability, Availability, Maintainability, Safety
RANS	Reynolds Averaged Navier-Stokes
RAT	Ram Air Turbine
RCE	Remote Component Environment

RPAS	Remotely Piloted Aerial System
RSM	Response Surface Model
RTO	Rejected Take Off
SE	Systems Engineering
SFC	Specific Fuel Consumption
SLOC	Source Lines of Code
SRM	Switched Reluctance Motor
SRS	Stakeholder Requirement Specification
SysML	System Modeling Language
SYSMOD	Systems Modeling Process
TAS	True Air Speed
TLAR	Top Level Aircraft Requirement
TODR	Take-Off Distance Required
TOFL	Take-Off Field Length
TRAS	Thrust Reverser Actuation System
TRU	Transformer Rectifier Unit
TTM	Time To Market
UAV	Unmanned Aerial Vehicle
UC	Use Case
UML	Unified Modeling Language
VCM	Vapor Cycle Machine
VLM	Vortex Lattice Method

VMS	Vehicle Management System
WER	Weight Estimation Relationship
WIPS	Wing Ice Protection System
XDSM	eXtended Design Structure Matrix

Chapter 1

Motivations and objectives of the research

1.1 Introduction

The aircraft design is a long and complex process. Thousands of employees are involved along the entire development[1] phase, which might last for several years, on average five [1]. Moreover, aircraft design is a multidisciplinary and collaborative process. Several design disciplines – i.e. domains of the aircraft design process – are indeed involved within the development of the new product. The main design disciplines encompass aerodynamics, structures, propulsion, on-board systems (equivalently referred as subsystems and aircraft systems), flight mechanics, performance, overall aircraft synthesis, emissions, costs estimation. All these disciplines often generate conflicts and contrasts between them, as their optimal solutions are not converging to a unique one. For instance, a thinner wing is required to reduce aerodynamic drag, but a thicker one is lighter [2]. Hence, a balance between the aerodynamic and structural design is needed to achieve the "best" solution, which results from the compromise between the two disciplines. To overcome – or at least to minimize – the conflicts among all the design

[1] The terms **design** and **development** are used as synonyms in the present dissertation.

disciplines, Multidisciplinary Design and Optimization (MDO) techniques are employed by aeronautical industries and research centers. As stated by Sobieski [3], MDO is "a methodology for design of complex engineering systems that are governed by mutually interacting physical phenomena and made up of distinct interacting subsystems (suitable for systems for which) in their design, everything influences everything else". Entering in the details of the MDO architectures, techniques and algorithms is out of the scope of the present dissertation, as detailed studies are collected in other works in literature, as in [2], [3], [4], [5]. In simple terms, the goal of a MDO problem is the optimization of a measure of merit – often named "design objective" – that generally is represented by the minimization of the aircraft development cost or by the minimization of the Maximum Take-Off Weight (MTOW). In addition, the design and optimization process is subjected to constraints, such as performance characteristics like the take-off distance requirement, and geometrical limits, for instance the maximum wingspan. The MDO techniques aim at overcoming all the boundaries among the disciplines, obtaining the global optimal solution, in accordance with all the design requirements.

In addition to this, new market demands are moving towards products with higher performance, more environmentally friendly and with lower operating costs (e.g. with lower fuel consumptions). Since a few years, the main aeronautical industries are developing technological improvements in several disciplines, for instance in:

- structures, with the increment of the percentage of composite materials, hence designing lighter aircraft. For example, the Boeing Company states that the high use of composite materials (about 50%) for the airframe and the primary structure of the Boeing 787 entails up to 20% of weight savings and a reduction of nearly 30% of the structures maintenance costs [6].
- propulsion, with the introduction of more efficient engines, as the new Pratt & Whitney PW1000G [7] installed aboard the Airbus A320 neo family, where significant benefits in terms of fuel consumption (up to -20% per seat by 2020), operating costs and engine noise and polluting NOx emissions (-50%) are claimed [8].
- aircraft subsystems, with the development of innovative on-board systems characterized by more efficient engines power off-takes. Innovative on-board system architectures are already adopted in civil aviation, as the case of the Boeing 787 and the Airbus A380 (see Section 1.2).

Moreover, since many years several research centers and universities are carrying on projects focused on the study and development of innovative concepts and disruptive technologies. Examples of these novel solutions are represented by the Blended Wing Body (BWB) aircraft ([9], [10], [11]), or the joined-wing aircraft ([12], [13], [14]). The main objectives of these two kinds of concept are the reduction of the aerodynamic drag or the decrease of the structural weight. Other studies are also focused on the hybrid-electric propulsion, as described in Section 1.2.

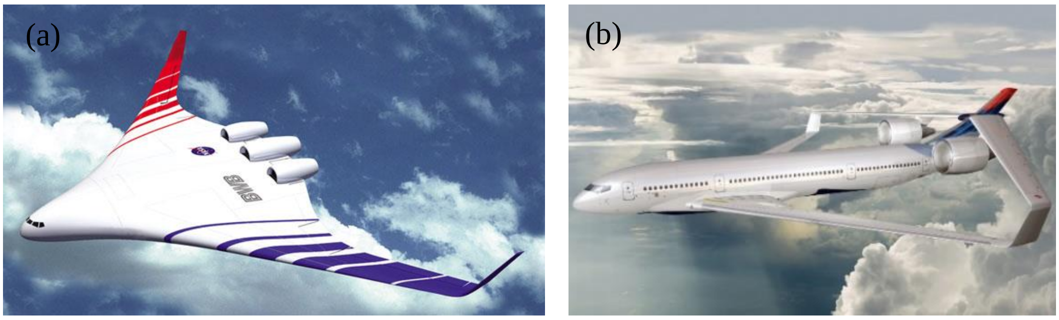


Figure 1: Two innovative concept studies: (a) the BWB and (b) the joined-wing aircraft [15].

However, a new innovative and disruptive product is not sufficient alone to guarantee the success of a new aircraft development program. Aeronautical companies have to deal with other competitors. Certainly, a competitive new airplane should be characterized by better performance in comparison with other aircraft belonging to the same class. Nonetheless, the success of a new aircraft depends on its price and its time of introduction into the market. Therefore, other important objectives of an aeronautical company are the decrease of the design and production costs, the reduction of the Time To Market (TTM), i.e. the time duration between the conception of a new product until its launch on the market, and the minimization of the risk of wrong decisions taken during the development of the new product. These wrong decisions might derive from wrong results obtained from the studies and the analysis performed during the earlier phases of the design process. The conceptual design phase is indeed characterized by a low knowledge of the future product. However, the designer has a large freedom about the design options and solutions (see Figure 2).

During the conceptual design phase many choices about the aircraft are taken, for instance the type of propulsion system of a general aviation airplane, making a selection between conventional or hybrid system. These preliminary decisions may be revealed as unfavorable, but only at the end of the design process, when higher fidelity tools and analysis codes are employed and more knowledge about the design is available. But, during the final stages of the aircraft development the design freedom is minimal, and making changes would be very expensive, causing also delays in the commercialization and delivery of the product. Therefore, it is mandatory to take proper decisions early in the design process, when the cost of changes is low.

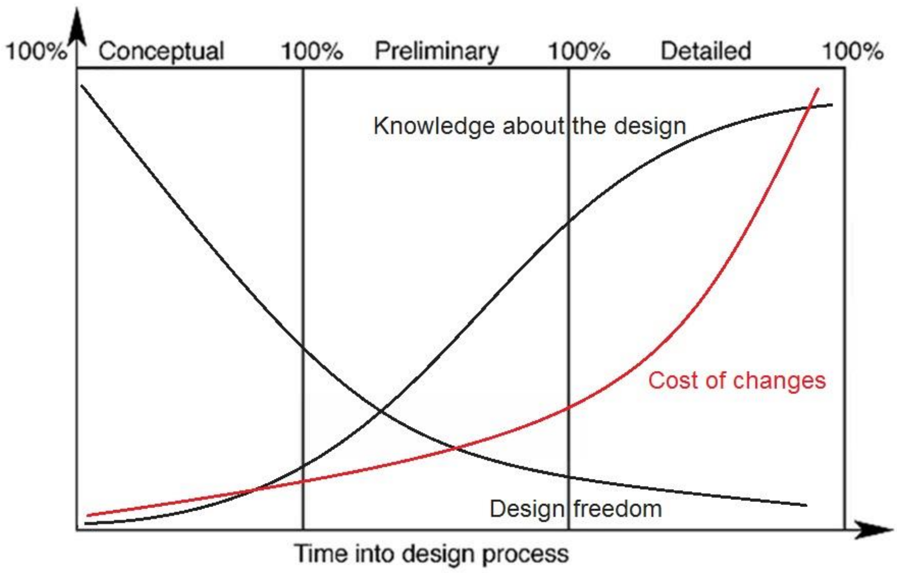


Figure 2: Knowledge about the design, design freedom and cost of changes during the aircraft development process (adapted from [5]).

1.2 Design of innovative on-board systems

As stated in the introduction, aircraft design is a multidisciplinary process. Several experts with different disciplinary competences (i.e. expertize, skills) are involved during the development of a new aeronautical product. Among all the design disciplines previously mentioned, on-board systems design is one with the highest impact on the overall aircraft. The pie charts represented in Figure 3 clearly show the impact of the subsystems on the overall aircraft, in terms of weight, cost and maintenance.

Other significant numbers are presented by Scholz. According to his studies, aircraft development and operating costs are affected by subsystems for about one third [16]. Moreover, the percentage of on-board systems mass on the overall aircraft empty weight ranges between a minimum of 23% – in case of modern long-range civil aircraft – to a maximum of 40%, considering smaller airplanes as business jets [17]. The aircraft mass is the specification mostly affected by on-board systems. However, important subsystems effects are observed on mission fuel quantity. A small percentage of the power or thrust generated by the propulsive system is indeed converted in secondary power. This means that a portion of the airflow passing through the stages of the engine compressors is typically bled, supplying hot and high pressure air to the aircraft pneumatic users, as the Environmental Control System (ECS) and the Ice Protection System (IPS). Furthermore, mechanical power is extracted from the engines by means of an Accessory Gear Box (AGB), moving electrical generators – hence transforming mechanical power in electric power – and hydraulic pumps, so supplying hydraulic users, for instance flight control actuators. The schema in Figure 4 depicts the traditional secondary power conversion [18]. The transformation of part of the propulsive power into secondary power "costs" part of the mission fuel. Generally, the fuel needed for the power supply of the on-board systems represents about 5% of the total mission fuel [19].

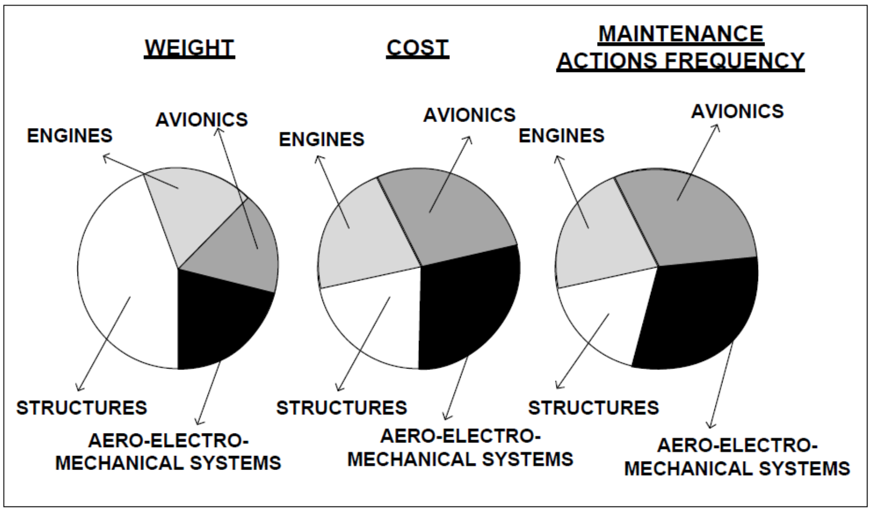


Figure 3: Impacts of on-board systems on the entire aircraft [20].

Another impact of the on-board systems is on the aerodynamic of the aircraft. Subsystem installation affects the aerodynamic drag and thus the airplane performance [21], [18]. Two simple examples could be given in support of this statement. The first one refers to the flap fairings, in which flaps kinematics are hosted [22]. The latter is represented by the shape of the belly of fuselage ahead the wings, which is usually affected by the installation of the Air Cycle Machines (ACMs).

Finally, aircraft subsystems have other relevant impacts, for instance on airplane center of gravity, volumes, safety and maintenance aspects. However, the most important effect of on-board systems may be on the Life Cycle Cost (LCC). Both acquisition and operating costs deeply depend on subsystems.

Due to the very important role of on-board systems on the entire aircraft level, and particularly in the case of unconventional subsystems, in the following Sections an overview of the state-of-the-art of two kinds of innovative subsystems is provided. The first one refers to the new trend of shifting from a conventional subsystems architecture – see the schema depicted in Figure 4 – to an All Electric architecture. The innovative All Electric configuration is characterized by the removal of the pneumatic and hydraulic power in favor of an increase of the electric power. The second advance of subsystem concerns the Hybrid-Electric Propulsion System (HEPS), which could represent a radical breakthrough in the aeronautical field. Both these two innovative solutions have the main aim of increasing the product efficiency, hence reducing fuel consumption and therefore operating costs.

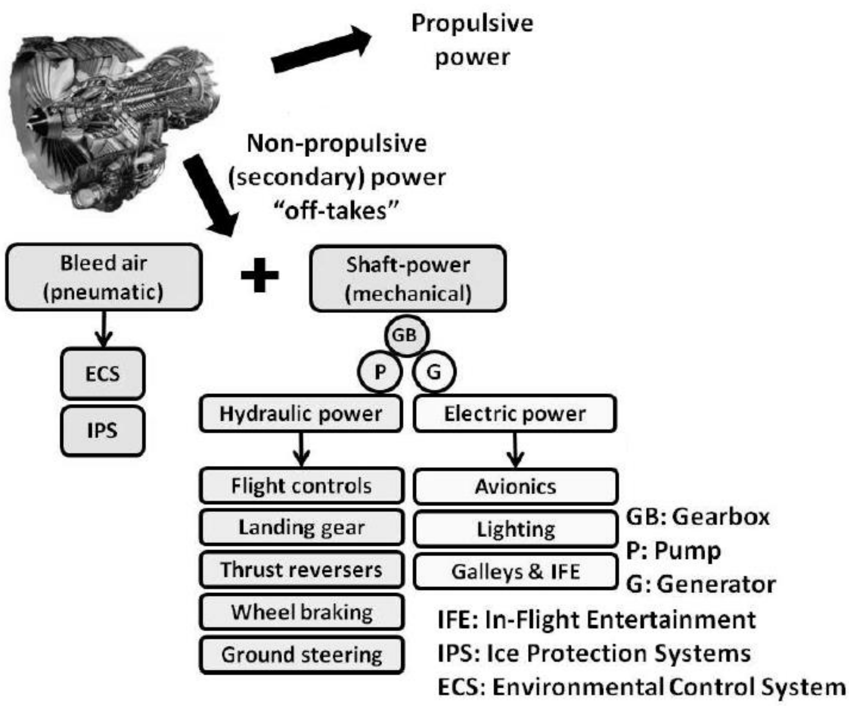


Figure 4: Transformation of secondary power – adapted from [18].

In conclusion, on the basis of all the subsystem impacts on the entire aircraft so far mentioned, it is worth noting that the study and sizing of on-board systems should be performed since a conceptual phase of the design process. Various are indeed the influences of subsystem results – as masses, power off-takes, costs – on the Overall Aircraft Design (OAD) process.

1.2.1 Conventional and More/All Electric subsystem architectures

As previously introduced and below depicted in Figure 5 (a), traditionally part of the power generated by the propulsion system (i.e. the engine) is converted in non-propulsive or secondary power to supply on-board systems. Three typologies of non-propulsive power are possible [23]:

- Electric power, which is transformed from mechanical power by means of generators mounted on the AGB.
- Hydraulic power, obtained from mechanical power through Engine Driven Pumps (EDPs). The EDPs are generally connected to the AGBs, but they could be supplied by electric motors or driven by air. Typical hydraulic users are actuators of the Flight Control System (FCS) and landing gear.
- Pneumatic power, consisting in high temperature and pressure airflow bled from the intermediate or high pressure engine compressor. State-of-the-art on-board system configurations employ pneumatic power for the air-conditioning and pressurization system and for the IPS. Furthermore, pressurized air runs a dedicated turbine to start the engines.

The weakest point of the conventional architecture is represented by the bleed air off-take, consisting in penalties affecting the performance of the engine. For a typical medium-sized passenger aircraft with a propulsive power of 40 MW, about 1.74 MW is transformed in secondary power, of which 1.2 MW is in the form of pneumatic power [24]. Then, it becomes clear that a more efficient solution is required. The schema of Figure 5 (b) represents an example of innovative subsystem solution. In this configuration, the mechanical power gathered from the engine shafts is transformed only in electric power. Electricity is then employed to supply electric users, which could encompass electric actuators installed in place of hydraulic ones. The bleed air off-take is removed. Consequently, the IPS is electrified – i.e. the anti-ice and de-ice systems consist in electrical resistances, as explained in [25] – and the airflow for the ECS is get from the external environment and pressurized by dedicated compressors moved by electric motors. The last kind of power, the hydraulic one, if needed, is obtained through Electric Motor Pumps (EMPs), which replace the EDPs.

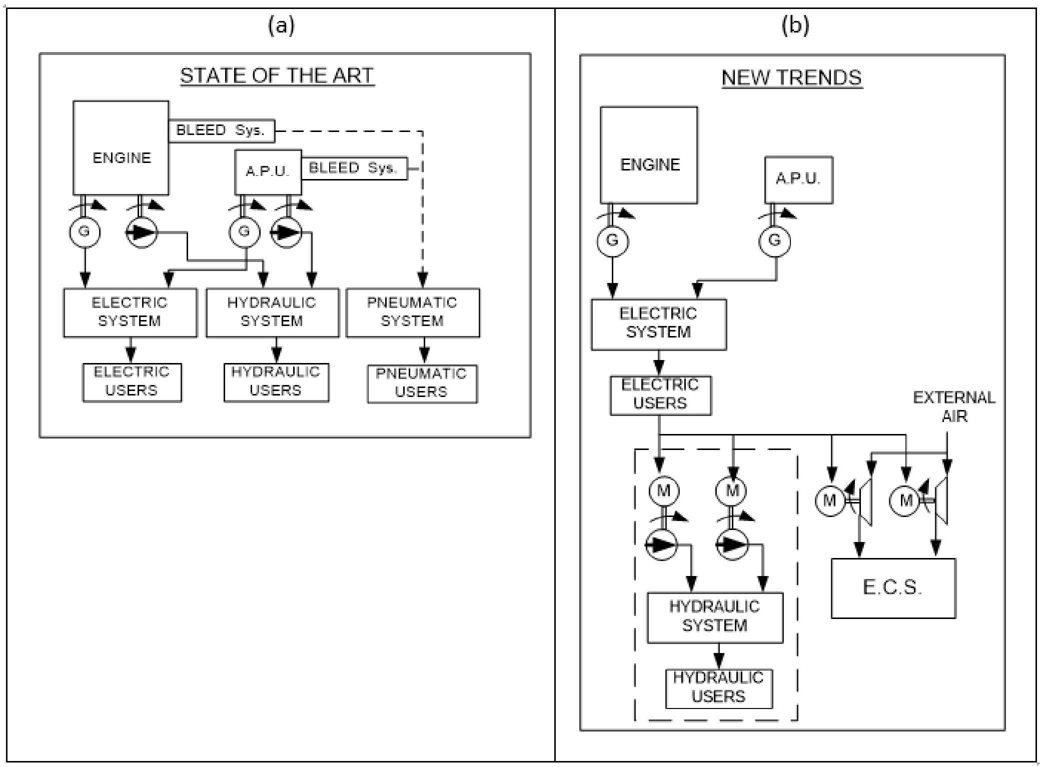


Figure 5: Schemas of (a) state-of-the-art and (b) More Electric systems architectures [26].

Even if the More Electric architecture – which is characterized by the removal of a type of secondary power (hydraulic or pneumatic) – and the All Electric architecture – an extreme version of the More Electric architecture in which only electric power is produced – represent an innovation, some current aircraft show these types of new advances.

Two examples are here proposed. The former refers to the Airbus A380. The innovative feature of this aircraft is represented by the FCS. For the first time in aviation history, Electro-Hydrostatic (EHAs) and Electro-Mechanical Actuators (EMAs) are installed aboard a civil passenger transport aircraft [27]. In more details, one of the three hydraulic circuits is replaced by two electrical lines. This solution is identified as “2H/2E” arrangement. Thus, almost every primary mobile surface is moved by a traditional hydraulic actuator and potentially by an EHA[2],

[2] On the Airbus A380, for safety reasons, two spoilers and the rudder are powered by Electrical Backup Hydraulic Actuators (EBHAs), which combine the features of EHAs and conventional hydraulic actuators.

which is set in stand-by mode. Leading and trailing edge surfaces are actuated by the hydraulic system and by EMAs. This innovative configuration has entailed around 450 kg in weight saving [28].

The second More Electric subsystems configuration is peculiar of the Boeing 787 ([29] and [30]). The Boeing 787 adopts a so-called “bleedless” configuration. Just a small portion of airflow is bled to protect the engine nacelles from the ice formation (Cowl Ice Protection System – CIPS). The air-conditioning and pressurization systems and the Wing Ice Protection System (WIPS) are supplied by the electric system. However, the “bleedless” configuration entails an increased electrical power generation. As shown in Figure 6, two 250 kVA generators per engine are installed. Taking into account the two 225 kVA generators mounted on the Auxiliary Power Unit (APU), the maximum hypothetical electric generated power reaches 1.45 MW. This value is very high compared to conventional systems, considering that the Boeing 767-400 generates 120 kVA per each one of its two engines. According to Boeing Company [30], this efficient solution entails a 3% of fuel saving per mission, against a significant increment of the subsystems weight.

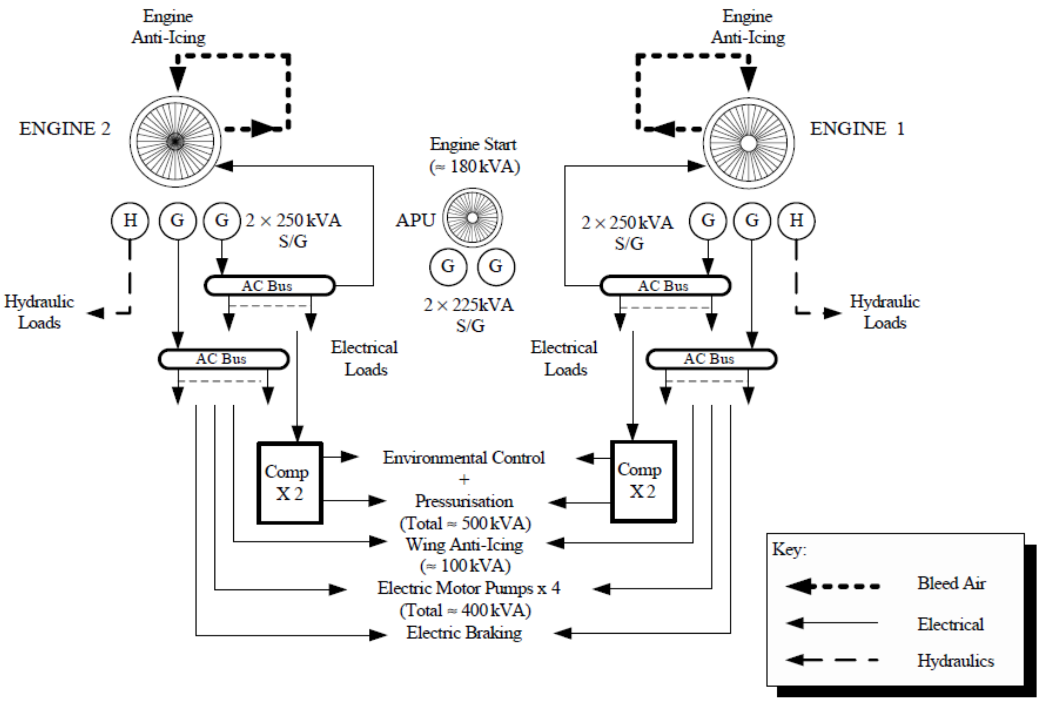


Figure 6: Boeing 787 electric loads [6].

1.2.2 Hybrid-electric propulsion system

A second type of innovative subsystem is represented by the HEPS. The aim of this technology is to produce more efficient propulsive power combining the endothermic source (i.e. the Internal Combustion Engine – ICE) and the electric source (i.e. the Electric Motor – EM) [31]. This kind of hybrid-electric propulsion is already consolidated in the automotive and nautical fields [32], pushed by the main advantages of reduction of fuel consumption and air pollution. However, this innovation in aeronautics is still premature. The weak point of the hybrid propulsion is indeed represented by the low energy density – meant as the amount of energy per unit of mass – of the electric accumulators. It is worth noting that the energy density of the AVGAS is about 44 MJ/kg, while the energy density of a lithium-ion battery could reach values only up to 0.6-0.8 MJ/Kg [33]. This means that heavy batteries could be installed aboard the airplane, affecting range and endurance of the aircraft.

While many aircraft with an all-electric propulsion have been designed, only a few examples of hybrid-electric concepts are worth mentioning. In 2011 Siemens, Diamond Aircraft Industries, and EADS presented at the Paris Air Show the first aircraft propelled by a HEPS, with the aim of demonstrating the feasibility of the hybrid technology in aeronautics [34]. This aircraft, a motor-glider named Diamond DA36 E-Star (Figure 7), is characterized by a propeller powered by an electric motor of 70 kW, which is supplied by both electric energy storage and a small 30 kW Wankel ICE linked to an electric generator.

Figure 7: Diamond DA36 E-Star [www.airliners.net].

This type of propulsion architecture is named series hybrid. A schema of the series hybrid architecture is depicted in Figure 8. As shown, the propeller is driven only by an electric motor, which is supplied by an electric generator connected to a thermal engine operating at a higher efficiency point, assisted by electric accumulators [35]. Advantages and disadvantages of this kind of hybrid architecture are listed in Table 1 [35].

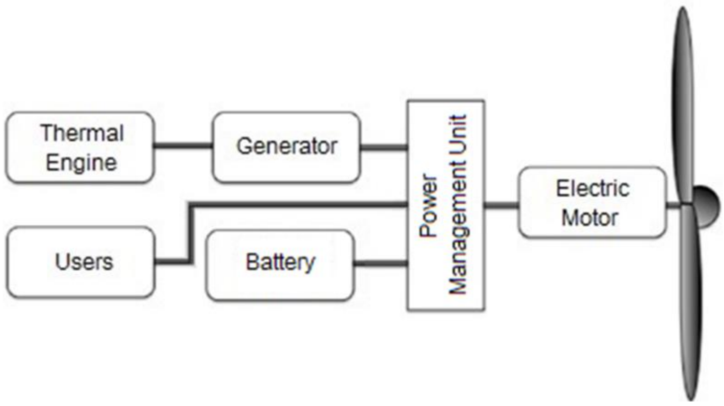


Figure 8: Schema of the series hybrid architecture.

Table 1: Pros and cons of the series hybrid architecture [35].

Pros:
1. The ICE operates with optimal conditions of torque and rotary speed, entailing maximum efficiency
2. The ICE operates at a nearly constant angular speed. Hence, it’s more reliable and it requires less maintenance
3. The “only-electric” mode is feasible
4. Batteries could be recharged during descent
Cons:
1. The electric motor is sized for the maximum power, with consequent weight increment
2. Reduction of the global efficiency due to the energy conversions

A second hybrid propulsion configuration is defined parallel architecture. The parallel hybrid is characterized by the mechanical coupling via a gearbox or a belt of the ICE with an electric moto-generator (see the schema in Figure 9). The thermal and electric machines both supply mechanical power to the propeller during the take-off and other flight mission phases in which extra-propulsive power is required. During other mission phases, such as cruise, the subsystem could operate in “traditional mode”. The propulsive power is generated by the thermal engine, which besides moves the electrical machine generating electrical secondary power. An additional operative mode of the parallel hybrid system is the “only-electric” one, which could be operated during the ground taxi.

Advantages and disadvantages of the parallel hybrid architecture are listed in Table 2 [36].

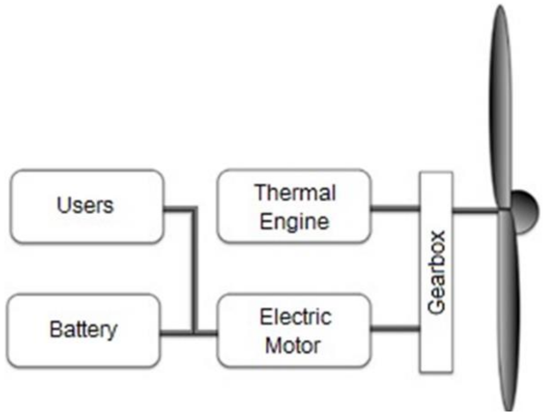


Figure 9: Schema of the parallel hybrid architecture.

Table 2: Pros and cons of the parallel hybrid architecture [36].

Pros:
1. *Powerboost* is supplied when peak power is required
2. The ICE is downsized, entailing weight and volume reductions
3. The safety level in case of ICE failure is augmented. The electric drive entails an increase of the gliding distance
4. Power is recovered during the descent, using the propeller as Ram Air Turbine (RAT). Batteries could be recharged
5. The “only-electric” mode is feasible, for instance during the ground taxi phase
Cons:
1. A mechanical clutch could be required to connect/disconnect the ICE

Several studies have been conducted focusing on this type of architecture, as will be described in subsection 1.2.1. Moreover, the potentialities of the parallel hybrid architecture have been proved through a test bench realized by Flight Design [37] (Figure 10).

The solution proposed by Flight Design is characterized by a fixed connection through a belt of a 115 hp (85.8 kW) Rotax 914 engine and a 30 kW Permanent Magnet electric motor-generator. During the take-off and climb phases, the EM could be employed as electric motor, supplied by a 130 V Lithium Iron Phosphate ($LiFePO_4$) battery pack. This type of energy storage is characterized by a good energy density (over 90 Wh/kg), entailing a total mass of nearly 30 kg. The power flow among the components is managed by an electronic controller, in which a control law based on the throttle position is implemented. According to this control law, for throttle values greater than 90%, both the ICE and the EM provide propulsive power ("hybrid" mode). Otherwise, the electrical machine is moved by the thermal engine, generating electric power ("traditional" mode). If the throttle is set between 20% and 85%, part of the generated electric energy is used for battery charging.

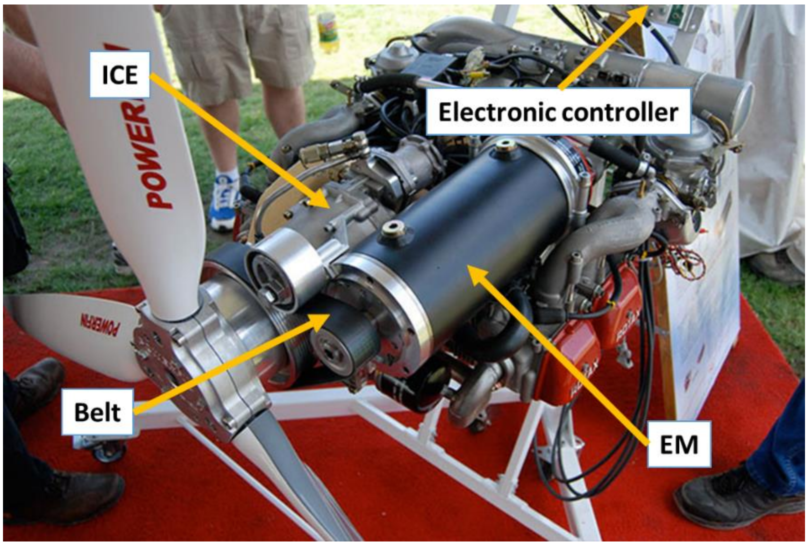


Figure 10: Flight Design's parallel hybrid bench (adapted from [www.wired.com]).

In conclusion of the present Section, it is worth mentioning the first aircraft powered by a parallel hybrid powertrain: the single-seat ultralight Alatus, conceived and realized at the University of Cambridge in 2010 [38]. This hybrid demonstrator is characterized by a 2.8 kW four-stroke thermal engine mechanically connected to a 12 kW brushless electric motor, which is supplied by Lithium-Polymer (LiPo) batteries with a capacity of 2.3 kWh.

1.3 Literature survey on current design methodologies

In Section 1.1, two main motivations at the base of the current doctoral research have been presented. The first motivation refers to the need of enhancement of the aircraft conceptual design phase, targeting the development of better and more competitive products, integrating all the involved design disciplines and reducing time and development costs. The entire aeronautical research community is addressing these objectives since the last decades. New approaches, methodologies and techniques are being developed, aided by the latest advances in Information Technology (IT), as growth of computational power and increment of the processing speed. A brief overview on the main research studies present in literature is presented in subsection 1.3.1.

The second motivation is relative to the new trend of products designed or studied, as improved current solutions or new disruptive concepts. In Section 1.2, some innovations regarding the aircraft subsystems have been presented, concerning specifically More and All Electric system architectures and Hybrid Propulsion Systems. Several research studies are being conducted on these new technologies, of which a survey is provided in subsection 1.3.2.

1.3.1 The enhancement of the aircraft design process

In his PhD dissertation [5], La Rocca claims that an improvement of the traditional aircraft design methodology is necessary to develop more technological and more competitive products. Moreover, La Rocca states that the most promising design methodology for the enhancement of the aircraft development process is the MDO approach. The same theory is endorsed by Raymer [2], who has conducted his doctoral research on the assessment of optimization methods evaluating their ability in finding the optimal solution and estimating their execution time. Within the context of his doctorate, optimization algorithms have been coded and implemented within a conceptual design framework to allow the automated re-design of the aircraft according to evolution of the design parameters during the MDO process. Moreover, MDO approaches entail a widening of the design space, allowing the investigation and development of novel and unconventional solutions [39]. The development of innovative concepts involves also the necessity of including all the design disciplines in an earlier phase of the design process and employing higher fidelity analysis tools, as design methods based on statistical data are only applicable for conventional solutions. Furthermore, higher fidelity tools can increase the level of confidence of the results, supporting the designers in making design decisions, relying on more affordable analyses [5].

The potentialities concerning the MDO technologies have encouraged the development of MDO frameworks, both within academic and industrial contexts. MDO frameworks are set up with the aim of integrating several disciplinary models, eventually defining the optimal solution. Several examples of MDO frameworks are present in literature. For instance, the Preliminary Aircraft Design and Optimization (PrADO) tool has been proposed [40]. PrADO encompasses several computer codes, each one concerning a specific aircraft design discipline. By means of this MDO framework, design analyses are conducted, to determine non-optimal solutions after convergences on design parameters. Additionally, sensitivity analyses based on the variation of design variables within the ranges specified by the user can be made. Eventually, PrADO allows the determination of optimal solutions thanks to the inclusion of optimization algorithms. One of the strengths of the proposed tool is its modular structure, which makes the framework extremely flexible for the application in different kinds of design and optimization problems, as the development of different types of unconventional configurations. Analogous to PrADO is another tool, called OpenMDAO, which is an open source Multidisciplinary Design Analysis and Optimization (MDAO) framework developed by Nasa ([41], [42], [43]). The publicly availability and the frequently updated documentation have made OpenMDAO one of the most popular MDO tools. Another well-known MDO framework is MICADO, developed by the RWTH Aachen University [44]. MICADO stands for Multidisciplinary Integrated Conceptual Aircraft Design and Optimization. This MDO tool allows both aircraft parametric studies and optimizations. MICADO requires a minimum of user inputs, mainly high level requirements and some specifications, and it quickly investigates a wide design space with low computational effort. Analogously, MDO technologies have been implemented within industrial environments (e.g. [45], [46], [47], [48]). However, industrial contexts are characterized by the interaction and collaboration among numerous experts belonging to large design teams. The automation of MDO workflows can accelerate the design process and increase productivity, but more interactive and intuitive processes are still needed to support the collaborative and iterative development process [49]. Other than technical issues – for instance limitations concerning computational power – an MDO process is indeed impeded by non-technical barriers, too. Belie [50] identifies non-technical barriers in:

- complexity inherent to the MDO problem itself, due to the complexity of the aeronautical product and its development;
- handling and management of a large quantity of data elements, which should be stored, accessed and assessed;
- difficult communication among several experts, belonging to different departments or institutions, with different culture, background and idiom;
- low confidence on the results obtained through the MDO framework, due to the high level of automation of the MDO processes.

Aimed at overcoming part of these non-technical barriers, a multilevel MDO framework is proposed by the Canadian aeronautical industry Bombardier [49]: different analysis tools with different levels of detail, and appropriate problem formulations and optimization strategies are defined for each stage of the entire development process, from the initial design phase to the most advanced one. The German Aerospace Center DLR instead has proposed a distributed MDO framework [51]. This design and optimization environment encompasses several disciplinary tools hosted in different DLR sites in Germany (hence the adjective "distributed"). All these codes are characterized by different levels of fidelity (see Section 2.5), and are developed and owned by different disciplinary teams, as the engineering knowledge is spread among the various sites. This solution aims at solving some of the barriers addressed by Belie. In this regard, a common data model named CPACS (Common Parametric Aircraft Configuration Schema) [52] allows the exchange of information between the disciplinary codes, facilitating the integration of the analysis tools and hence the assembly of the workflow, and managing all the generated data. CPASC is an xml file serving as a central description of the aircraft, in terms of properties and geometry. This schema is employed by the disciplinary experts for the extraction of the inputs needed by the disciplinary modules. The outputs obtained through the analyses are then stored inside the CPACS file. In this way, the exchange of information among disciplines is considerably enhanced, and the number of interfaces is radically reduced.

However, a connection among disciplinary tool is not sufficient alone to deploy more competitive and faster MDO processes. All the competences of the several involved experts should be combined to create a collaborative other than multidisciplinary MDO framework [53]. Several projects (e.g. VIVACE [54], CESAR [55], CRESCENDO [56] and TOICA [57]) are focused on the overcoming of collaboration obstacles, deriving processes and techniques aimed at easing and enhancing the integration of the different disciplines. One of these projects, the European Horizon 2020 AGILE project [56], is devoted to the conception and exploitation of techniques and processes for the improvement and acceleration of the aircraft collaborative design and optimization. In particular, AGILE is targeting the realization of a new generation MDO framework, as will be described in Section 2.4. An overview of the AGILE framework will be instead presented in Section 3.5.2.

Other approaches might be adopted to achieve the needs stated in Section 1.1. According to La Rocca, a Knowledge Based Engineering (KBE) approach combined to MDO methodologies can be adopted to reduce design time and development costs. The KBE enhances the aircraft design by reusing product and process results, automating repetitive and non-creative design tasks, and employing MDO techniques in all aircraft design phases. Although KBE might be adopted in different engineering field as automotive ([58], [59]), software engineering [60], and aerospace [61], this approach still lacks of interest within the academic community. A possible cause of this shortage of spread of KBE might lie in its employment exclusively within a few automotive and aerospace industries, without fostering scientific research on it.

Differently from the KBE, other approaches are deeply spreading within the academic research community. One of these approaches is the Systems Engineering (SE), which aims at improving the design process – i.e. reducing development time and costs and minimizing design risks due to wrong choices – focusing on all the involved design disciplines and on the integration and interconnection among them [62]. As will be presented later in more details (Section 2.2), the SE approach pays great attention on the product requirements and demanded functionalities since the earliest phases of the design process, but targeting also the entire Life Cycle. Furthermore, the SE tends to include all the involved stakeholders within the development, without neglecting to assess in which political, social and economic context the product will be operated [63]. In this way, the risks due to wrong design choices are reduced, as much emphasis is given on all the requirements of the whole product. The SE approach aims at disseminating a model-based approach – i.e. the Model Based Systems Engineering (MBSE) – to enhance the collaborative design among different experts by means of the exchange of information through models instead of documents. The MBSE is supported by the System Modeling Language (SysML), a standard graphical modeling language used to describe the behavior of the designed product, which nevertheless does not produce any analytical result [64]. Therefore, an incompatibility among these models and the disciplinary models employed in an MDO framework is present. Studies and attempts for the integration among the MBSE and the MDO tools are present in literature ([64], [65]). In particular, the integration of MBSE and MDO techniques is envisaged for the widening of the design space, with the aim of identifying unconventional solutions, entailing a higher number of trade-off analyses. However, a development methodology that recommends how to join the MBSE and the MDO approaches is still missing.

1.3.2 On-board systems design methodologies

The current subsection presents a brief review of the studies and research conducted by the scientific community about the development of the aircraft on-board systems. In particular, the attention is posed on the methods and techniques proposed for the development of innovative subsystems, mainly More and All Electric architectures and hybrid propulsion systems. In general, researchers are in agreement that a deeper on-board systems design should be moved up to an earliest phase of the design phase (e.g. [21], [66] and [67]). This means that the traditional subsystems design methodologies (e.g. [68], [69] and [70]) should be overcome by more accurate (e.g. physics-based) algorithms. Traditional methodologies are indeed based on statistical data, therefore effective only for conventional architectures. Moreover, these methods are limited to the preliminary estimation of the subsystem masses. Innovative architectures instead might entail advantages at aircraft level, for instance decrease of fuel consumption due to the reduction power off-takes. Therefore, the secondary power demanded by on-board systems should be properly evaluated from the early beginning of the design process. Furthermore, the design of the on-board systems should be carried out within an OAD context, in order to evaluate all the effects of the subsystems development at the aircraft level, as variation of aircraft masses, fuel consumption, and aerodynamic drag.

The traditional on-board systems design process is structured on the Air Transport Association (ATA) subsystems breakdown [71]. The ATA distinguishes 32 subsystems (ATA chapters), providing a standard separation among aircraft systems and characterizing the subsystem development process. However, the ATA breakdown results inadequate for the definition of innovative subsystem architectures. Therefore, Liscouët-Hanke *et al.* remark the necessity of overcoming the traditional ATA breakdown, establishing a new methodology able to foster the conception of innovative architectures [72]. Functions-driven approaches are indeed proposed in [21] and [66] with the aim of enlarging and deeply investigating the design space, assessing novel subsystem architectures. This functional development should bring to the definition of the physical elements required to provide some functionalities. A more exhaustive functional-induction based methodology centered on the definition of innovative subsystem architectures is proposed in [73] and [74]. The authors identify two types of functions: boundary and induced. The boundary functions derive from the high level requirements, and entail the selection of physical components. Subsequently, physical components involve further induced functions, which bring to the definition of additional physical elements. Thanks to this proposed methodology, the complete set of system functions can be defined, identify the best (innovative) architecture able to fulfil all the functionalities demanded by the stakeholders. However, the advantages given by a functional approach can be strengthened by means of a more formalized methodology. A more structured approach is represented by the previously mentioned MBSE, which by employment of standard diagrams (e.g. the SysML diagrams) provides the designer with guidelines and procedures for the system functional development. Several methodologies have been proposed over the years with the aim of supporting the MBSE. For example, the Object–Oriented Systems Engineering Methodology (OOSEM) [75] adopts the SysML language to support the specification, analysis, design and verification of new products. This methodology focuses on the definition of product goals, mission and operative scenarios, identifying all the stakeholders and all their needs. Analogously, a method developed by the Jet Propulsion Laboratory (JPL) [76] entails the determination of a model and state-based architecture. The states represent the conditions reached by the product during the mission, while the evolution of the product states is described by the models. Another methodology is proposed in [77]: the SYSMOD Systems Modeling Process. This methodology includes the following activities: identification of the stakeholders, requirements elicitation, definition of the system context, requirements analysis and definition system architecture. Other methodologies are available in literature, as [78], [79], [80] and [81]. One of these methodologies – the IBM Harmony methodology [82] – has the advantage of being associated to proprietary software tools.

Other studies present in literature are indeed focused on the development of algorithms for the preliminary sizing of the on-board systems. As previously stated, the traditional design methodologies are supported by large databases collecting data of conventional airplanes. Therefore, new algorithms should be developed for the design of both conventional and innovative subsystem architectures. Furthermore, due to the required employment of these algorithms in an early phase of the development process, models for subsystem sizing should be based on aircraft parameters available at the beginning of the design. For instance, a methodology for the initial estimation of the electric power required by aircraft subsystems all along the mission profile is described in [83]. The methodology proposed by the authors is based on high level requirements (e.g. number of passengers), and aircraft results obtained after the first design converged iterations, as cabin volume, maximum fuel weight and wing area. Historical subsystem data is exploited for the determination of fitting equations built on appropriate parameters. However, the statistical basis might make the proposed methodology unsuitable for certain novel subsystem architectures, for which few public data is available. Instead, original algorithms for the preliminary estimation of masses and required power off-takes of conventional and also innovative subsystems are reported in [21]. These models concern the following subsystems: WIPS, Commercial Cabin System (CCS), Pneumatic Power Generation and Distribution System (PPGDS) and Electric Power Generation and Distribution System (EPGDS). The algorithms have been realized within a collaboration with Airbus, therefore based on restricted data and partially published. Other algorithms are proposed by Lammering [66]. The models tackle the preliminary estimation of only the secondary power demanded by ECS, IPS and FCS. Analogously, in [67] are proposed algorithms for the preliminary computation of power off-takes demanded by conventional and innovative on-board systems. Other than the subsystems targeted in [66], the authors propose equations for the design of landing gear, EPGDS and HPGDS. More detailed studies concerning only the FCS are presented in [84] and [85]. Both the publications deal with the design of innovative actuators, i.e. EMA, EHA and a hybrid actuation system. However, the proposed algorithms might result too much detailed in a subsystem preliminary design, but can be employed for deeper studies at component-level. Algorithms for the preliminary design of on-board systems are presented in the doctoral dissertation of Chakraborty [86]. Contrary to the other previously cited researchers, the author describes design models for additional subsystems. Other than the FCS, ECS, IPS, PPGDS, EPGDS and HPGDS, the proposed algorithms target the preliminary mass and secondary power estimation of the landing gear system, the electric taxi system, the Thrust Reverser Actuation System (TRAS) and the Mechanical Power Generation and Distribution System (MPGDS). The proposed models efficiently provide the conceptual development of the on-board systems, although several inputs of the proposed models might be too much detailed or unavailable in a conceptual design phase.

A common interest of many researchers working on subsystems design regards the evaluation of the on-board system influences on the entire aircraft design process. Different types of subsystem architectures – from conventional to All Electric – impact differently the aircraft level. New airplanes with lower MTOW or fuel weight can be designed thanks to a proper selection of the subsystem architecture. However, the subsystem architecture must be selected during the conceptual design phase, when low knowledge about the new product is available. Therefore, the aircraft subsystems design discipline should be integrated within an OAD process, investigating the impacts of on-board systems on the other design disciplines since the earliest phase of the development. A simulation framework realized to analyze the effects of subsystem architectures at aircraft level and vice-versa is described in [72]. The framework proposed by the authors couples subsystem models to an aircraft performance model. The performance model evaluates the impact of subsystem results – i.e. masses, power off-takes, drag – on the overall aircraft performance all along the mission profile. The framework is indeed designed to compute the thrust required by the propulsion system and thus the fuel consumed during the mission. An analogous framework in proposed by Lammering [66], in which however more emphasis is put on the multidisciplinary aspect of the aircraft design process. In this case, on-board systems design models are implemented within MICADO, besides encompassing several disciplines involved in the OAD process, i.e. weight estimation, aerodynamics, flight mechanics, engine performance and cost estimation. Regarding in particular the engine performance, the authors employ the tool GasTurb ([87], [88]). GasTurb is a commercial software aimed at the simulation of on/off design conditions of several engine decks. However, the automatic re-design of the engine on the basis of the design parameters varying at each iteration is limited. Chakraborty *et al.* instead propose an OAD integrated to subsystem design framework with a different solution [18]. The fuel consumption due to shaft power off-takes is indeed computed through the method proposed in [89]. The method of SAE AIR 1168/8 [90] is indeed implemented for the estimation of the fuel consumed by bleed air off-takes. In this way, variations of subsystem architectures and re-designs of the propulsion system within and automated design framework are favored.

Part of the current survey is devoted to the presentation of the main research studies conducted on one of the most innovative subsystems: the hybrid propulsion system. Since the last years, many research studies have been conducted on this new kind of technology. Other than the prototypes and test benches previously described (see subsection 1.2.2), it is worth providing a brief review on the work present in literature about this new aircraft system. Due to the advantages listed in Table 2, the attention in the current dissertation is focused on the parallel architecture. The parallel architecture can be more favorable than the serial one as it might entail a lighter propulsion system. Based on the available literature, the majority of publications deal with the hybridization of aircraft powered by piston engines. However, studies conducted on other types of propulsion system are also present, as concerning turboprop aircraft ([91]) and jet airplanes ([92]). In general, published papers are mainly focused on the design and simulation of the components of the hybrid system. In [36] and [93] a simulation model of the hybrid propulsion system integrated with an aircraft performance model is presented. An Unmanned Aerial Vehicle (UAV) is chosen as case study. The aim of this model is the evaluation of aircraft characteristics – as airspeed, fuel flow, attitude, position and altitude – according to the simulated behavior of the hybrid propulsion system. Analogously, a hybrid aircraft simulation model is proposed by Hung *et al.* [35], again referred to an unmanned airplane. A more detailed analysis on the equipment of a HEPS is presented in [94]. Furthermore, the authors associate the subsystem model with an extremely simplified model of the airplane, to evaluate performance characteristics as thrust and aircraft speed. Other than simulation models, test benches have been realized, too. Apart from the Flight Design's prototype previously introduced (subsection 1.2.2), scale-down test benches have been constructed, as those described in [36] and [94], both dealing with parallel hybrid systems for UAVs. However, a general design methodology for the conceptual sizing of the parallel hybrid propulsion system and assessment of the impacts at aircraft level is missing in literature. A few design equations are presented by Harmon *et al.* [95], but their study is not generic but centered on the design of an unmanned aircraft. Nevertheless, the paper claims potential advantages in terms of energy savings due to the introduction of the hybrid technology.

The present literature survey ends with a brief presentation on the application studies performed by other researchers concerning the preliminary design of More and All Electric aircraft and hybrid airplanes. A detailed overview on the studies and research projects carried out in the context of the electrification of the conventional on-board system until the beginning of 2000s is provided by Jones [96]. In particular, it is worth mentioning the NASA's Integrated Digital Electric Aircraft (IDEA) project, aimed at assessing the potential advantages of an All Electric version of the Boeing 767. From the published report [97], it results that the electrification of the reference aircraft entails a MTOW reduction of nearly 2.62% and a fuel saving of about 4.78%. Similar results are obtained by Chakraborty *et al.* with regard to a short range transport aircraft (Airbus A320 or Boeing 737). The authors claim a MTOW reduction of the All Electric version of about 2.92%, while the fuel weight decrease exceeds 5%. Different percentages are instead computed by Lammering [66], at least regarding the quantity of fuel, where a decrement of 9-10% is envisaged in the case of an Airbus A320 characterized by the All Electric architecture. Concerning the MTOW, its reduction is in line with the literature (~2-3%). Concerning the preliminary design of hybrid aircraft, the study proposed by Bagassi *et al.* worth to be mentioned [98]. In this work, a comparison in terms of performance, fuel consumption and aircraft weights between two configurations of the same airplane is presented. The former is characterized by the conventional propulsion system, while the latter concerns a hybrid electric system. However, from the comparison is not evident which one of the two solutions is more convenient, as aircraft weights and airplane requirements (e.g. payload and range) result different.

1.4 Main objectives of the research and structure of the thesis

The previous Section presented a literature survey on research studies addressing the same motivations of the present doctoral dissertation, stated in Section 1.1. The following considerations emerge from the literature survey:

- collaborative MDO environments are currently being developed (e.g. H2020 AGILE project). However, the integration of the on-board system design discipline within collaborative MDO environments is still missing;
- although works are present in literature regarding the evaluation of the impact of subsystem at aircraft level during the conceptual design phase, these works are still few. More research projects should be carried out targeting this aim. Furthermore, all the studies found in literature mainly deal with conventional, More and All Electric subsystem architectures. Other on-board system technologies should be investigated as well, as the hybrid propulsion system;
- methodologies aimed at enhancing the MDO within a collaborative environment are currently being developed, e.g. the AGILE framework development process (see Chapter 3). However, these methodologies might be improved by integrating techniques aimed at focusing more on customer's needs, enhancing the exchange of product information (e.g. requirements, specifications, results) among experts, containing design costs;
- several studies address the functional analysis for the initial definition of subsystem architectures. In this context, the MBSE approach targets a preliminary determination of the subsystem architecture based on the

elicitation and development of functional requirements. Several researchers are studying the integration of the MBSE within the MDO processes, but this integration is hampered by the incompatibility of the methods and the employed tools;

- few algorithms are present in literature for the preliminary sizing of conventional and innovative subsystems, at least in terms of masses and power off-takes. Furthermore, some of these algorithms are based on restricted data and therefore not public. Moreover, other algorithms require parameters that are too much detailed for a conceptual design phase. In addition, algorithms for the preliminary sizing of the parallel hybrid propulsion system are still missing in literature;
- few works deal with the preliminary design of aircraft equipped with More and All Electric subsystem architectures and hybrid propulsion systems. More investigations regarding these innovative solutions should be conducted.

Therefore, two main objectives of the current dissertation derive from the two main motivations of the research and the limitations of the current studies available in literature. The first main objective is the analysis and development of design techniques for the aircraft systems conceptual design to be included within a collaborative and multidisciplinary development methodology. These techniques would guide the integration of the subsystems design discipline within the MDO environment from the earliest phase of the development process. A five-step methodology based on a SE approach is employed. This methodology starts from the elicitation of the high level aircraft requirements and subsystem required functionalities, and it allows the definition of a MDO problem required for the determination of the optimal solution.

The second main objective is relative to the application of the proposed design methodology. Four case studies are considered, each one concerning the application of some techniques, approaches and steps of the proposed methodology. In particular, these case studies are focused on the design and optimization of two kinds of subsystems. The first one is relative to the hybrid-electric propulsion system, while the second type is relative to conventional and innovative on-board system architectures.

In order to reach the top objectives stated above, the present dissertation is organized as follows. In Chapter 2, some key elements at the base of the proposed methodology are presented. In particular, an overview of the SE and the Design-

To-Cost approaches is provided. Then, the state-of-the-art and the evolution of the MDO design frameworks are described. Chapter 2 ends with a swift presentation of system models.

The entire MDO methodology is presented in Chapter 3. The presented methodology derives from the AGILE development process. Design techniques supporting the aircraft multidisciplinary and multi-experts design and optimization process are investigated, developed and integrated within the MDO methodology.

The proposed methodology is then employed for the design and optimization of hybrid-electric propulsion systems and conventional and innovative on-board system architectures (Chapter 4). The obtained outcomes show the validity, effectiveness and the potentialities of the proposed methodology.

Finally, Chapter 5 concludes the dissertation, presenting the main original contributes of the research, but also highlighting the limitations of the work. All these limitation will be motivation for further research.

Chapter 2

A Systems Engineering approach in a collaborative design framework

2.1 Introduction

From the previous Chapter it emerged the need of enhancing the current design methodologies in order to develop new and successful aircraft in a multidisciplinary and multi-experts collaborative environment. In the present dissertation, a methodology based on the SE approach is employed, which will be presented in Chapter 3. This methodology is being developed within the context of AGILE project ([99], [100]), and it focuses on the involvement of design experts and on the integration and interconnection among their design disciplines. In particular, the original contribution of the present doctoral dissertation consists in the assessment and inclusion of design techniques aimed at enhancing the integration of the subsystems design discipline within this collaborative MDO methodology. The first part of the present Chapter therefore gives a brief overview on this approach.

As the methodology described in the dissertation shall enable the collaboration of several disciplinary experts, one of its aims should be the setup of a collaborative and multidisciplinary design framework, i.e. an aircraft development environment connecting and linking together all the available disciplinary analyses (e.g. codes, tools owned by different experts) required to solve the MDO problem. The evolution of the state-of-the-art multidisciplinary design frameworks is one of the topics of the current Chapter. The following Chapter – which is devoted to the description of the design methodology – will present how setting up a design framework based on the SE approach.

Another approach employed within the current research activities and described below is the Design-To-Cost (DTC). This approach aims at reducing the product development and production costs, in agreement with the objectives stated in the introductory Chapter about the reduction of the product LCC.

The current Chapter ends with the presentation of a few types of models, employed in both functional and performance designs. The functional designs are aimed at developing the system on the basis of the functional requirements. According to NASA [63], functional requirements define the functions necessary to accomplish the objectives. Instead, performance designs are driven by performance requirements, which define how well the functions are required to be performed [63]. In particular, the conclusive Section of the present Chapter remarks the importance of modeling during the system design process.

2.2 A Systems Engineering approach

SE is a design approach capable of integrating all the design disciplines, overcoming all the conflicts and constraints among them. According to the SE discipline, the attention is focused on the system. The system is defined as an ensemble of elements combined together in order to satisfy a common objective [101], [102]. The interrelated components interacting together consist of persons, organizations, procedures, software, equipment and facilities [63]. Within a SE perspective, the attention during the design process is indeed focused on the entire whole system, without considering its parts alone. Moreover, the SE approach gives great importance to the definition and the development of the stakeholder requirements. This fact is evident from the definition of SE given by the International Council on Systems Engineering (INCOSE). INCOSE defines the SE as:

> "*an interdisciplinary approach and means to enable the realization of successful systems. It focuses on defining customer needs and required functionality early in the development cycle, documenting requirements, and then proceeding with design synthesis and system validation while considering the complete problem. Systems Engineering considers both the business and the technical needs of all*

customers with the goal of providing a quality product that meets the user needs" [101].

In other words, SE is a methodical approach to design, create and operate systems during the whole life cycle, from a conceptual design phase up to the retirement of the product, passing through all the stages of the product life cycle. Furthermore, with a SE approach all the agents involved within the design and their relative needs and requirements are considered. Their involvement is very important for the success of a project. According to a report of the Standish Group [103], almost one program every two fails due to scarce requirements, insufficient involvement of stakeholders and bad management of the development process (see Table 3). Moreover, SE is useful for the design of complex systems as well as innovative and/or disruptive products. In this case, the knowledge about the future product is extremely low, and without adopting a methodical approach as the SE the probability of project failure is very high.

Table 3: Typical causes of project failures [103].

Incomplete requirements	13,1%
Lack of user involvement	12,4%
Lack of resources	10,6%
Unrealistic expectations	9,9%
Lack of executive support	9,3%
Changing requirements/specifications	8,7%
Lack of planning	8,1%
Didn't need it any longer	7,5%

In order to meet all the stakeholders' needs and requirements, decision gates during all the life cycle stages are defined. This gates represent milestones of the development process, during which reviews of the project are made. These audits aim at deciding whether the project is mature enough to continue towards the following life stage. Several life cycle models are defined in literature, according to different manufacturers. Figure 11 depicts a comparison among some life cycles, with indicated the typical project decision gates.

However, the main aircraft design references ([68], [104], [70]) divide the development process in three stages: the conceptual, the preliminary and the detailed design phases. This kind of distinction among development stages is used as reference in the current dissertation. During the conceptual design phase only a little percentage (about 1%) of the experts involved in the project participates. This design stage has a time duration of several days or a few weeks, and it is aimed at generating as many as possible concepts on the basis of the high level requirements, assessing and comparing them, selecting at the end the baseline, characterized by the 3-views and some preliminary data. The selected baseline is then refined during the preliminary design phase, which involves more experts (about 9% of the total) and lasts for some months. During this stage, only few details might be changed. The majority of experts is involved during the last design phase. This stage is characterized by a duration of the order of years, and it is aimed at designing the entire airplane down to every single detail.

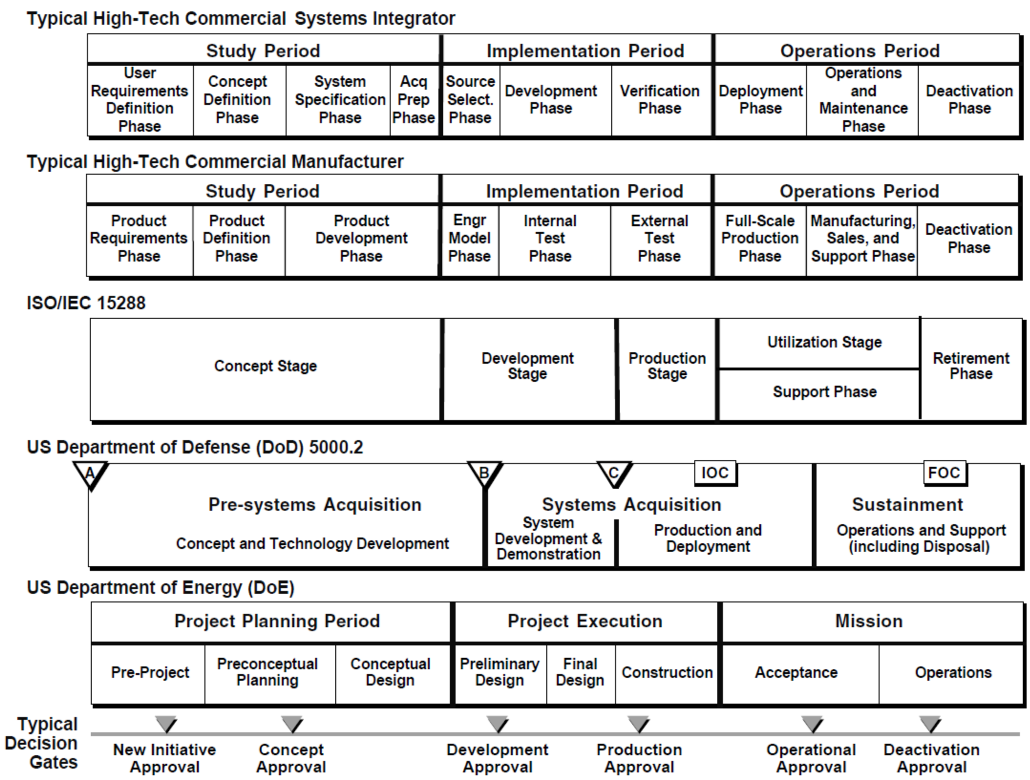


Figure 11: Comparison of life cycle models [101].

All the main SE activities performed during the life cycle are typically illustrated in a so-called “V” model [62] (Figure 12). This model is a graphical representation of the system life cycle, and it characterized by a “V” shape, hence its denomination. In the “V” model, time and project maturity proceed from left to right. The left side of the “V” depicts a top-down design from the agreement on stakeholders’ requirements to the derivation of system specifications, from system to components level. The right side of the “V” deals with integration, verification and validation activities, back from components to system level.

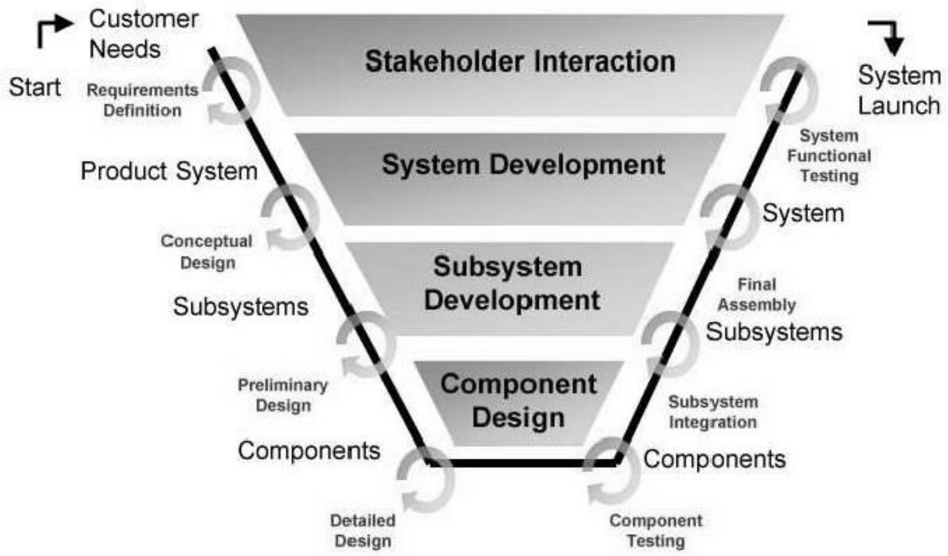


Figure 12: "V" model.

The current dissertation is focused on the left side of the "V". In particular, the proposed design methodology starts from the stakeholders' requirements definition and encompasses the entire conceptual design phase and part of the preliminary development stage.

The requirements of a new product typically encompass system performance characteristics and design schedule constraints. Therefore, the design choices are indeed taken to meet all the performance and schedule goals. Hence, the development and production costs are results of the design process [70]. Certainly, costs might be monitored during the whole development process, but interventions to reduce costs are minor. This approach might entail to high design and production costs, involving low profits or causing a noncompetitive product. A DTC approach alters this design philosophy, identifying the final product cost as a requirement to be complied. The DTC will be object of the following Section.

2.3 The Design-To-Cost

Requirements as TTM, performance characteristics and technology typically drive the development process, while the "cost" criteria might be a result instead of a main design target. Thus, the selling price of a new product is established from the costs of the development and the production phases. This approach might be advantageous only if the product is superior to its competitors. Otherwise, the high costs would entail low profits or a high product price, causing low selling rates and hence a failure of the program. In some companies the "cost" criteria is more important, but actions required to lower it are usually done later in the design process, often resulting inadequate.

The DTC is a cost management philosophy aimed at considering the total development and production cost as an objective of the design of a new product. This cost is a target of the design, similarly to all the other requirements, as the TTM, the performance, the technology level. The U.S. Department of Defense defines the DTC as:

> "*an acquisition management technique to achieve defense system designs that meet stated cost requirements. Cost is addressed on a continuing basis as part of a system's development* and *production process. The technique embodies early establishment of realistic but rigorous cost targets and a determined effort to achieve them"* [105].

Therefore, the DTC strategy states cost objectives at the beginning of the development process. Several trade-off studies are then conducted with the goal of designing according to the predetermined cost objectives. The DTC approach follows a step-by-step procedure described hereafter.

1) <u>Formulation of the target price requirement.</u>

   The target price of the new product derives from the willingness to pay of the customer, in the case of a military user, or from a civilian market survey, analyzing products with similar characteristics and performance. The formulation of the target price requirement is an important step of the DCT approach. A too high price might cause a scarce product demand, while a too low estimation would entail a reduced profit margin.

2) <u>Determination of the target cost.</u>

   The target cost of the new product is derived from the target price, from which the following cost items are subtracted: profit and risk margins, and overhead spending, which includes general costs as tools, rents and assurances.

3) <u>Target cost allocation to the system parts.</u>

   The determined cost is divided and allocated among the subsystems and components of the product. This target cost breakdown involves a team of experts coming from several disciplines of the company, as Engineering, Finance, Manufacturing, Sales and Supplier Management. Each expert is responsible for the own target cost.

4) Design and costs estimation.

Next step refers to the cost estimation of each subsystem and component of the Product Breakdown Structure (PBS). In this stage, several alternative solutions are designed and compared, with the aim of selecting a baseline compliant with the target costs. Methodologies as the Value Engineering (see Section 3.7) may be adopted to define and evaluate the basic product functions and capabilities required by the customer. Every team is responsible for the design in accordance with the predetermined cost requirements. All the cost items are taken into account, as production models of the product during the whole life cycle are required, in order to support the decision-making process. An overview of the main LCC estimation models is given at the end of this Section.

5) Trade-off analyses and selection of the baseline.

Once several alternative solutions and relative costs have been derived, trade-off analyses are conducted with the aim of selecting the baseline, always considering the cost as a requirement.

6) Costs monitoring.

Periodically during the development process, the cost estimations are compared with the target costs. This task is performed by a Cost Review Board, which encompasses representatives of Program Management, Chief Engineering, Marketing and Sales. The customer is excluded from these cost reviews.

7) Corrective actions for the costs reduction.

In case the costs are not compliant with the target costs, corrective actions should be immediately made to reduce them.

2.3.1 Models for costs estimation

The DTC is based on an estimation of costs and relative risks since a conceptual phase of the design and during the entire development process. However, the earliest design phases are characterized by a low knowledge about the product.

Hence, accurate estimations are not possible at these stages. This knowledge increases during the development process, and consequently the precision and the accuracy of the estimation augment. Thus, different methodologies for costs estimation are required in accordance with each phase the design process. The Association for Advancement of Cost Engineering (AACE) identifies indeed five classes of costs estimation [106], on the basis of the level of project definition (see Figure 13). Within each one of these classes different cost models are employed, with different purposes. All these estimation models can be divided in four categories:

- Analogy;
- Parametric or statistical;
- Bottoms up or engineering;
- Actual costs.

	Primary Characteristic	*Secondary Characteristic*		
ESTIMATE CLASS	**LEVEL OF PROJECT DEFINITION** Expressed as % of complete definition	**END USAGE** Typical purpose of estimate	**METHODOLOGY** Typical estimating method	**PREPARATION EFFORT** Typical degree effort relative to least cost index of 1[b]
Class 5	0% to 2%	Concept Screening	Capacity Factored, Parametric Models, Judgment, or Analogy	1
Class 4	1% to 15%	Study or Feasibility	Equipment Factored or Parametric Models	2 to 4
Class 3	10% to 40%	Budget, Authorization, or Control	Semi-Detailed Unit Costs with Assembly Level Line Items	3 to 10
Class 2	30% to 70%	Control or Bid/ Tender	Detailed Unit Cost with Forced Detailed Take-Off	4 to 20
Class 1	50% to 100%	Check Estimate or Bid/Tender	Detailed Unit Cost with Detailed Take-Off	5 to 100

Figure 13: AACE's cost estimation classes [106].

Each costs estimation method is characterized by advantages but also by limitations. Their use depends on the level of knowledge about the design and by the phase of the development process. Furthermore, all the methods might be more efficient whether data and results concerning past programs are available, as this information might be use as comparisons.

Short descriptions of the four estimation methods are hereby provided ([107], [108]). For each approach, potentialities and limits are highlighted.

The first method – "Analogy" – is based on a database in which similar products with relative specifications and costs are collected. Certainly, this approach could be applied only whether a correlation exits between the product under design and the past ones. In order to adequately estimate new products, complexity factors are included within the "Analogy" method. These factors are expressed as percentages, and they are used to define the differences among old and new projects. Moreover, this approach requires low effort and short time. However, the statistical basis represents the weakest point of the method. This approach might be inadequate in case of innovative concepts, and it would bring to subjective choices during the quantification of the complexity factors.

The "Parametric or Statistical" cost model is based on cost divers deriving from an analysis of product dimensions, characteristics and performance. The cost drivers are parameters present in semi-empirical formulas named Cost Estimation Relationships (CERs). The reliability and the consistency of the CERs depend on the size of the statistical basis. This fact may represent a drawback of the method, as cost estimation of newer concepts would be unfeasible. Moreover, more time and more information in comparison with the "Analogy" approach are required. Notwithstanding, if the database collects numerous statistical points and the product is traditional, the cost results are highly reliable. Furthermore, the most considerable benefit regards the possibility of evaluating the effects on costs of modifications of the design.

A more detailed approach is the "Bottom-up" method. However, this method requires more precise and exhaustive data about the program, but this information is generally available during later phases of the design process. This method estimates all the man-hours necessary for the fulfillment of each process of the program and for the realization of every component of the product. Other cost items are considered, as indirect costs, raw materials, tools. Thus, the "Bottom-up" approach is employed in advanced phases of the design.

The last costs estimation method is based on the actual costs of the prototype or of the first realized product. Thanks to this method the estimation is highly accurate, but this approach can be employed only in the latest phases of the development process, when consistent changes of the design are impossible.

2.4 A collaborative design framework

The multidisciplinary feature of the aircraft design process requires the collaboration and integration of several disciplinary experts. As introduced in Section 1.3.1, MDO frameworks aim at connecting together all the disciplines involved in the development process. Connecting all the disciplines doesn't mean only linking all the simulation and design models, but also combining all the competences, which is hampered by non-technical barriers, as large data handling, interpretation of the results, communication among different disciplinary experts [50].

As already stated, the AGILE consortium is currently working on the definition of an innovative MDO framework, the so-called 3rd Generation MDO framework. The transition from the 1st to the 3rd Generation of MDO frameworks is described in [109] and the relative schemas are depicted in Figure 14.

In the 1st Generation MDO framework all the disciplinary modules and the optimizer are integrated within a unique monolithic system, i.e. inside a single computer or server. The design team has access to all the analysis modules. The main drawback of this architecture is represented by the lack of flexibility to modify or update the integrated analysis modules, both in case of improvement of the design modules and when the MDO framework's architecture should be modified to develop new configurations. Furthermore, this solution becomes unfeasible when more disciplines and effects are included. However, the 1st Generation MDO framework is currently employed in two scenarios. The former refers to high-fidelity design and analysis problems. The latter is relative to the conceptual design of a new aircraft, done through very simple models, with the aim of quickly investigating the effects deriving from the integration of a limited number of disciplines. An example of monolithic MDO architecture will be given in subsection 3.5.1, while results of a case study will be presented in Section 4.3.

In the 2nd Generation MDO framework, disciplinary modules are distributed on different dedicated computational facilities and they are owned by different experts affiliated to different departments or institutions. All these modules are called by a centralized design and optimization process monitored by the design lead team. However, the high level of automation of this kind of framework and the generation of a large quantity of analysis data entail confusion and skepticism on the results.

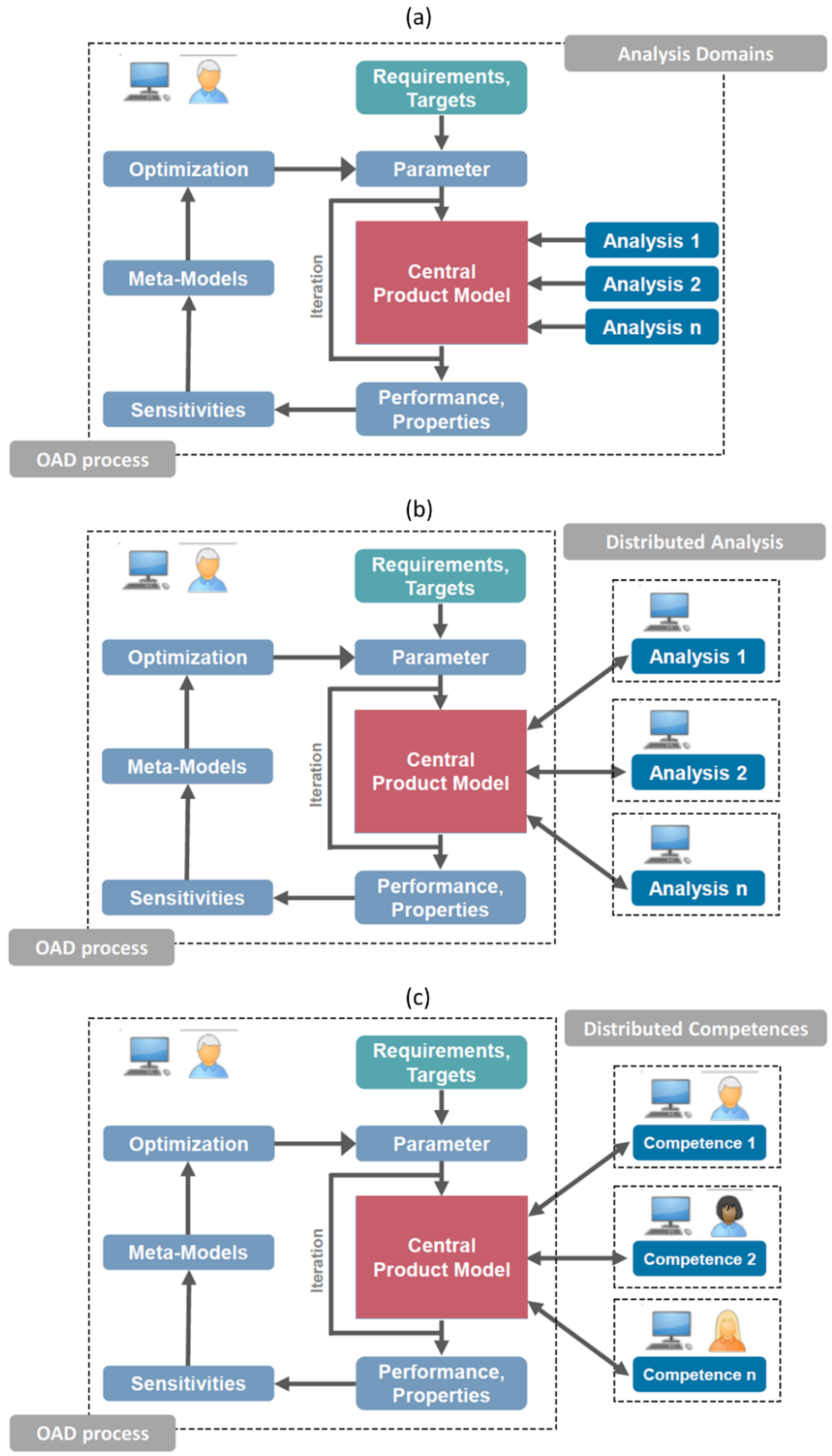


Figure 14: Generations of MDO frameworks: 1st (a), 2nd (b) and 3rd (c) generation [109].

The involvement of disciplinary experts is required for the monitoring of the design process and the assessment and validation of their own codes.

Therefore, an evolution of the 2nd Generation MDO architecture is represented by the 3rd Generation MDO framework. More than a connection of different tools, this most innovative kind of MDO framework includes technologies aimed at overcoming non-technical barriers concerning communication among different experts, with different skills, education, idiom and culture. Moreover, the 3rd Generation MDO framework targets the increment of confidence on the results obtained through models owned by different institutions, of which only the inputs and the outputs are public, while the algorithms are secret. The AGILE consortium has indeed developed technologies for the enhancement of the collaboration among different disciplinary experts. An example of the AGILE distributed architecture with particular focus on the on-board system design integration is provided in subsection 3.5.2. Some results of a second application study are presented in Section 4.4.

2.5 The importance of modeling in systems design

In the last decades, Model & Simulation (M&S) based techniques have acquired always more and more importance and relevance within the context of aircraft design [110]. The development process has moved from a “design-build-test” approach to a “simulate-test-simulate build” philosophy. This fact has been supported by the enhancement of IT systems, and by their increase in terms of computational power, speed and storage capacity. The main advantages of M&S approach encompass strong reductions in development costs, effort and design times.

A model might be both a functional and a physical representation of a system or part of it. A model is used to understand the behavior and to specify the structure of a system before its realization. In fact, through a model it is possible to simulate the system to verify that all the requirements are compliant.

A system model can be characterized by different fidelity levels, depending on the purpose of the model and the expected results. Four fidelity levels can be identified [111]:

- L0: based on empirical or statistic rules, can be used for the exploration of the design space.

- L1: based on simplified physics principles, can be adopted to model and analyze a limited amount of effects.
- L2: the physical behavior is represented by more accurate relationships and equations.
- L3: characterized by a very high computational costs, can be used to investigate only local effects, without exploring the whole design space.

Each design discipline is characterized by own fidelity levels, as shown in Table 4 for sake of example.

According to their purpose, system models can be categorized into several classes. In the current dissertation the attention is focused on functional and performance models.

Functional models are aimed at representing the logic behavior of the system. This models are employed in the very beginning of the design process, with the main goal of preliminarily defining a system architecture – i.e. an ensemble of components integrated together – able to be compliant with a set of functional requirements, as will be discussed in subsection 3.2.1 in more details. An example of functional requirement might be: “the aircraft shall autonomously land”. Therefore, functional models are employed to derive and develop all the system functionalities, defining at the end a “functional mock-up” of the system itself, which will be later sized from a performance perspective. Proceeding with the previously mentioned example, the aircraft shall have equipment able to detect its position during landing and to perform maneuvers to autonomously land.

Table 4: Example of fidelity levels for aircraft design disciplines [111].

Level	**Aerodynamics**	**Structures**	**Propulsion**	**On-board Systems**
L0	Empirical performance estimation	Handbook masses estimation	Simplified functions	Empirical mass estimation
L1	Linear analysis (VLM, Panel method)	Linear simplified models (beam theory)	Performance with generalized components	Steady state performance of main systems
L2	Nonlinear analysis (Euler)	Linear detailed models (FEM shells)	Performance real engine parameters	Transient state performance of whole subsystems
L3	Nonlinear non automated (RANS)	Nonlinear local analysis (buckling)	Detailed analysis (CFD)	Multidisciplinary simulation for critical system components

The System functional modeling can be performed by means of the SysML. The SysML derives from the UML (Unified Modeling Language) standard, and it is used for the conceptual development of complex systems through defined semantics [101]. The SysML exploits standard diagrams for the specification of system requirements, behavior, structure and parametric relationships. The nine SysML diagrams are schematized in Figure 15. The Requirement Diagram collects all the system's requirements, their hierarchies and their relationships. The remaining diagrams are split between two categories: behavior and structure diagrams. The former include Use Case (UC), Activity, Sequence and Statechart diagrams. The system's high level functionalities are represented in the Use Case Diagram. The Activity Diagram depicts the flow of all the activities performed by the system. The Sequence Diagram lists all the actions and messages exchanged by the collaborating parts of the system. The Statechart Diagram shows the state of the system and the events necessary to the transition among states. The latter category encompasses the Internal Block Diagram (IBD), the Block Definition Diagram (BDD) and the Package and Parametric diagrams. The IBD depicts the internal structure of the system, representing its parts, ports, and connectors. The BDD shows all the physical elements that compose the system. The Package Diagram represents the model from an organizational perspective, depicting the Packages containing elements of the model. The Parametric Diagram shows the parameters of the system, as performance, mass and other properties. The Figure 16 provides a brief overview on some of the SysML diagrams. More details are present in [82].

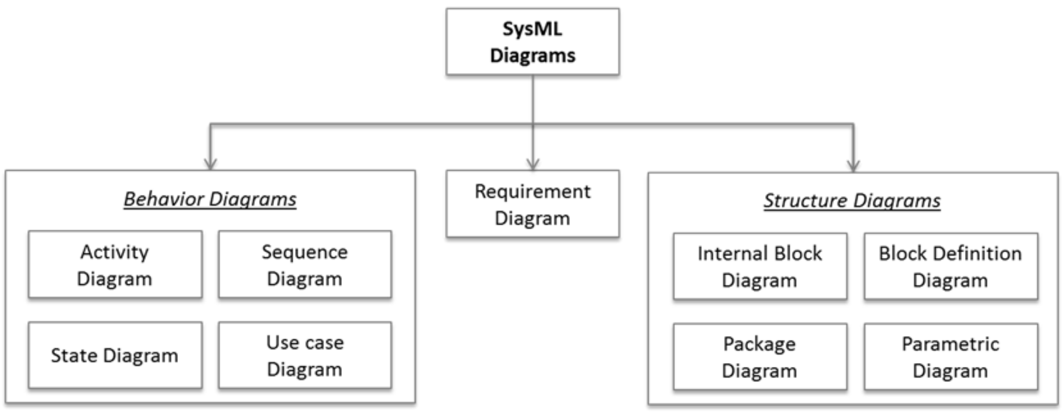


Figure 15: SysML diagrams: schematic.

The performance models instead include a set of equations defining the performance characteristics of the system. In other words, this kind of models is employed to size each part of the system according to the performance requirements. A performance required might be, for instance, "the aircraft shall

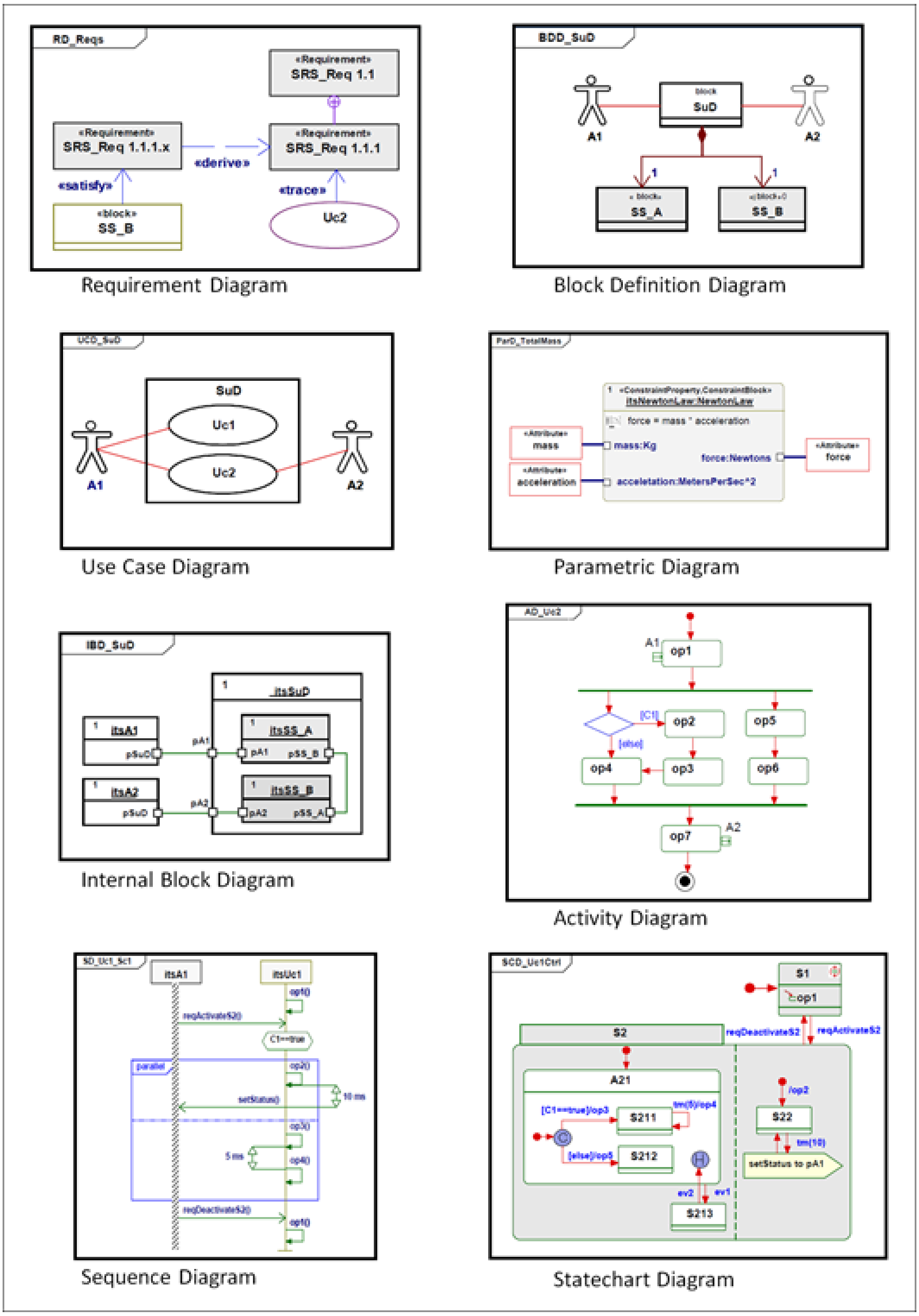


Figure 16: SysML diagrams: overview (adapted from [82]).

have a maximum Take-Off Field Length (TOFL) of 1500 m”. Therefore, all the system elements – as propulsion system, wing aerodynamics, wing area – should be designed in accordance with this requirement. The system model is subject to certain inputs named design variables or design factors, of which the domain of existence is named design space. A design variable is a parameter that can be freely controlled by a designer or by an optimizer. The wing AR is an example of design variable. Unfeasible conditions of the design space are instead known as design constraints. The system responses – i.e. the analysis outputs – are named state variables. The equations defining the relationships among design and state variables are implemented within a discipline analysis model. Different discipline analysis models simulate the different behaviors of the system, as the aerodynamics and the structural design. As the aircraft design is a multidisciplinary process, several discipline analyses should be integrated. Their responses might be exchanged among them. In this case, the disciplinary outputs are known as coupling variables. The estimated quantity of on-board systems power off-takes is for instance considered a coupling variable, as it affects the turbine engine efficiency and it is hence required as input by the propulsion discipline. Some state variables might be interest of the analysis. Generally, these results are monitored during each design iteration in order to define the optimal solution. These state variables are called objectives of the design process. For instance an objective of the analysis might be the minimization of the MTOW. A design analysis (hereafter defined as design problem) could be characterized by a single or multi-objective optimization. In the first case, a unique design objective is considered, while in the second case a compromise among the objectives shall be defined in order to select the “best” design solution.

Often, the disciplinary analyses might be conducted through the employment of the so-called Response Surface Models (RSMs). The RSMs – also defined as surrogate models or meta-models – are simplified versions of the disciplinary models. In other words, RSMs contain empirical and statistical equations derived by the multiple executions of the original model. These equations are independent from the physical laws at the base of the disciplines. The mathematical relations among inputs and outputs of the surrogate models are often in the form of low-order polynomials, generally belonging to the first or the second order. The main advantage of this kinds of model is their high execution velocity. High-fidelity disciplinary models might employ hours, days or even weeks before deriving a solution. RSMs instead are much faster. However, RSMs are only approximations of the true relationships. The level of approximation depends on the methodology employed to construct the RSM. Despite the approximations, the responses of the surrogate models might be considered acceptable, depending on their purpose. For instance, when several iterative runs must be performed to reach a converged solution, the process duration might be reduced adopting the employment of the RSM at least during the first iterations. Additionally, by means of the surrogate model, the designer might acquire a first idea of the behavior of a disciplinary analysis subject to a set of inputs.

As previously stated, the disciplinary RSMs are built performing several executions of the disciplinary models, varying at each iteration certain design variables within predetermined ranges. Based on a more mathematical definition, the design space is investigated by means of the techniques named Design Of Experiments (DOE) [112] (see Section 3.4). Each experiment is characterized by a different point of the design space in which the response shall be evaluated. From the selection of these experiments depends the accuracy of approximation and the cost – meant for instance as computational time – of the response surface. Several methods are available for the sampling of the experiments. All these methods are deterministic, as their samples – i.e. inputs – are placed in the design space according to predefined N-dimensional geometries. During early stages of the development process, screening experiments can be performed to preliminarily investigate the design space. For this aim, a Full Factorial Design approach [113] can be adopted to analyze all the possible combinations of design variables. Other examples of sampling methods are Orthogonal Arrays [114], Central Composite Design [115], Latin Hypercube Designs [116], Audze-Eglais' approach [117], Van Keulen's approach [118].

Chapter 3

Integration of subsystems design in a collaborative MDO process

3.1 A collaborative design methodology

The present Chapter describes a methodology for the collaborative and multidisciplinary aircraft design process. This collaborative design methodology has been developed within the AGILE project ([99], [100]). In particular, some techniques are proposed for the integration of the on-board systems design discipline within a collaborative MDO context. The methodology is aimed at defining a product architecture based on all the stakeholders' requirements, enhancing the collaboration among the design disciplines and selecting the optimal solution. Therefore, it follows a SE approach, as it starts from the elicitation and development of all the stakeholders' requirements, and it combines together all the main aircraft design disciplines, considering their relations and impacts among each other.

An overview of the collaborative design methodology is given below. In the following Sections it will be presented which design techniques are integrated in the AGILE development process and how they can enhance the aircraft subsystems design. In particular, the design techniques analyzed and proposed in the current dissertation are:

1) Functional analysis by means of a MBSE approach, aimed at defining a preliminary product architecture compliant with the behavior and capabilities expected by all the involved stakeholders.
2) Design of Experiments, aimed at investigating the design space.
3) Fuzzy Logic, for the multi-objective optimization and negotiation of part of the high level requirements of the aircraft.
4) Set up of a 1st Generation MDO framework, for the preliminary investigation of subsystems design effects on OAD and vice-versa, prior to the execution of a collaborative 3rd Generation MDO framework.
5) Design-To-Cost approach coupled with the Value Engineering method, with the aim of reducing the product cost removing all the product functionalities not required by the customer.

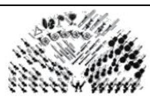
Aim of the development process is the setup of the MDO framework, starting from the high level aircraft requirements, deriving and developing all the stakeholders' requirements, collecting and integrating all the available disciplinary tools and competences, formulating the design and optimization problem, implementing the tool-chain and eventually obtaining the solution.

Therefore, this development process consists of five steps [100]:

- Step 1: Define design case and requirements
- Step 2: Specify complete and consistent data model and competence
- Step 3: Formulate design problem and solution strategy
- Step 4: Implement and verify the design framework
- Step 5: Create and select the design solution

A 3rd Generation MDO framework might encompass hundreds of experts, with different disciplinary competences and with different tasks. Each person involved within the development process is responsible of certain tasks and activities [53]. Other than the customers who require certain product capabilities, some people are in charge of setting up and controlling the overall process, both drafting and architecting the development process and enabling it through IT infrastructures. Other persons are more focused on the own discipline and other people are responsible for the improvement of the collaborative and multidisciplinary design process. Therefore, in [119] five MDO types of participant are identified, which are reported in Table 5. These five agents participate during the various phases of the MDO process performing dedicated roles. In order to have a clear understanding of the role of these five agents, comparisons with a symphony orchestra ensemble are done. In the description of the five steps, the tasks of each agent are presented.

Table 5: Participative agents of the collaborative MDO framework (adapted from [119]).

Participative Agents	**Responsibility**	**Orchestra Analogy**
Customer	Customer and primary user of the framework. Responsible for defining the design task, top level requirements, and available development lead-time.	Audience
Architect	Responsible for specification of the design case in the AGILE framework, such as collecting the required competences, defining the design phases and the dimensionality of the design space to be explored.	 Composer
Integrator	Responsible for the deployment of the design and optimization (sub-) processes, and for the management of such processes within the framework	 Conductor
Disciplinary expert	Responsible for providing design competence within the framework, such as a simulation for a specific discipline, or an optimization service.	 Performers
Collaborative expert	Responsible for providing the integration within the framework, necessary to connect the various disciplines and making them accessible to the framework.	 Ensemble

3.2 Step 1: Define design case and requirements

In this first step of the development process the specific design case and the different Top Level Aircraft Requirements (TLARs) that should be satisfied are defined by the customer. Eventually, the high level requirements are formalized and developed, deriving requirements at lower system levels.

As discussed in Section 1.1, one of the typical main causes of a project failure is the user's lack of satisfaction about the designed product, as its functionalities, performance characteristics and specifications are not agreeing with his expectations. In other words, a comprehensive elicitation of all the system requirements might be missing, due to the insufficient involvement of all the stakeholders. This gap between what the system users want and how the product is realized must be reduced since the very beginning of the development process. An approach based on the functional analysis can be adopted in this phase. By means of a functional analysis approach, all requirements at aircraft level can be captured and developed towards lower levels, as system and component levels [120]. Moreover, the functional approach leads to a definition of the product architecture free from the actual technology level, hence allowing a deeper investigation of the design space for the determination of innovative solutions [121].

In the current dissertation, a MBSE approach is adopted for the definition and derivation of the comprehensive list of system and subsystem requirements. The MBSE approach follows formalized methodologies for the capture and development at various levels of the product requirements, eventually deriving a functional architecture compliant with the developed functional requirements. This architecture will be then developed in the following steps of the design methodology. Moreover, a MBSE approach enhances the collaborative process as it allows a sharing of data and information – e.g. requirements, specifications – through standardized diagrams and models.

The definition of the design case and the elicitation and development of the system requirements are tasks performed by the architect in collaboration with the customer.

3.2.1 The aircraft functional design

As previously introduced, the functional design refers to the conceptual development of a new product from a functional perspective. In other words, the attention of the design of the new product is centered on its behavior, i.e. on what the system should accomplish, without focusing on its performance [63]. The purpose of the functional design is a first definition of a product architecture based on the capabilities required by the stakeholders. A system architecture is composed by several components integrated together with a global aim [74]. The functional development of this kind of system would analyze all the functions necessary to accomplish the capabilities required by the customer, hence bringing to the definition of one or more subsystem architectures. A trade-off analysis among all the architectural solutions would eventually bring to the selection of the baseline.

Conventional aircraft systems are characterized by standardized configurations and architectures. For this type of systems the functional design might be perceived as unnecessary. Different is the case of innovative products. The architecture of innovative subsystems is indeed not known *a priori*, but it derives from a functional development.

As already stated before, the here proposed methodology for the functional design of a (innovative) subsystem is based on a MBSE approach. INCOSE defines the MBSE as:

> *"the formalized application of modeling to support system requirements, design, analysis, verification, and validation activities beginning in the conceptual design phase and continuing throughout development and later life cycle phases"* [62].

As previously introduced, the MBSE differs from a "document-based" approach, as all the system requirements and specifications are collected by means of models instead of text documents, reports and drawings. Several are the benefits of the modeling activities compared to the "document-based" approach:

- All the complexities of the system are easily managed by the employment of models. With a "document-based" methodology it might be difficult to trace all the links and connections among the design analyses. Thus, all the effects of a potential design modification on the entire system would be evaluated through a MBSE approach.
- All the decisions taken during the design process are collected and documented.
- In general, the collaboration among different people is enhanced, as the information is shared through models instead of documents.
- The resulting design and analysis models could be adapted and exploited in future projects or programs, hence entailing reductions in working time and effort. Furthermore, several variants of a same baseline could be easily derived, studied and compared.
- The system might be simulated, at least from a functional-behavioral point of view. This would be necessary to verify whether the system is compliant with all the functional requirements. Furthermore, the simulation would be used as a demonstration towards the customer for his validation of the product behavior.

Among all the development methodologies supporting the MBSE approach (see subsection 1.3.2), the IBM Harmony [82] can be employed for the functional design of the new system thanks to its association to proprietary software tools. Following the IBM Harmony methodology, the designer is led in the employment of the SysML diagrams during the conceptual design of the system.

As shown in Figure 17, the IBM Harmony methodology is composed by three parts, as three are its objectives:

1) Identify and derive the capabilities of the system → Requirement Analysis

2) Identify the states and the functionalities of the system → System Functional Analysis
3) Define the system architecture based on its functionalities → Design Synthesis

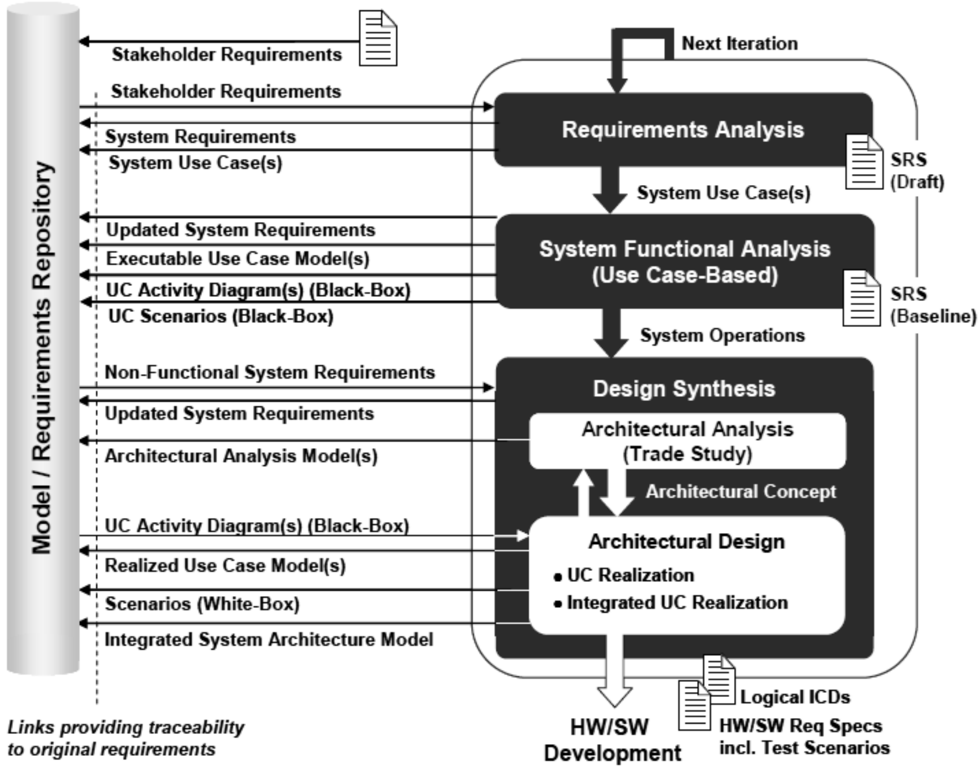


Figure 17: Phases of the IBM Harmony methodology [82].

In the following subsections, more details about the three phases of the IBM Harmony methodology represented in Figure 17 are provided. A case study of functional design applied to the development of a HEPS installed aboard an unmanned airplane will be described in Section 4.2.

Requirement analysis

The aim of the first phase of the IBM Harmony methodology is the analysis of the requirements. The stakeholders' requirements are analyzed and transformed into System Requirements, defining what the system shall do (functional requirements) and the performance characteristics of the system (performance requirements).

The “Requirement Analysis” phase starts with the analysis of the stakeholders’ requirements and the definition of the “Stakeholder Requirement Specification” (SRS), which is focused on the functionalities (or system capabilities) required by the stakeholders. Then, all these requirements are transformed in Requirement System Functions and documented in a draft version of the “System Requirements Specification”.

Later, the System Use Cases are defined. The UC describes a specific behavior of the system, as perceived by the actors, which could be persons, other systems or pieces of hardware. The UC also shows the relations and messages among the system and the external actors.

System Functional Analysis

In the second part of the methodology, the System Functional Requirements are transformed in a coherent description of the System Functions. This task is performed for each UC previously defined.

In the “System Functional Analysis” phase the behavior of the system is investigated by means of the Activity, Sequence and Statechart SysML diagrams.

Three are the possible alternatives of order of employment of these diagrams. The here chosen alternative is useful when the requirements should be developed. This approach starts with the definition of the UC functional flow (Black box Activity Diagram). From these functional flows, the UC scenarios (Black box Sequence Diagram) are derived, with the following creation of ports and interfaces. Finally, the behavior – in terms of states – of the system is modeled in the Statechart Diagram. Once the UC models and the relative functional requirements have been verified, a “Rainy Day Analysis” is required, looking for errors and behaviors of the system not covered in previous analyses.

Design Synthesis

The objective of the “Design Synthesis” phase is the development of the physical architecture able to satisfy the required functions, in compliance with the performance constraints. This last phase of the IBM Harmony methodology is composed by three groups of activity: the “Architectural Analysis”, the “Architectural Design” and the “Detailed Architectural Design”.

Generally, the “Design Synthesis” starts with an Architectural Analysis (Trade Study) as more hardware and software architectures compliant with a set of functional and performance requirements could exist. The aim of the Architectural Analysis is the definition of how the system should be realized in order to perform the functions previously defined. Therefore, among all the possible solutions compliant with the defined behavior of the system, the optimal one is identified, in accordance with all the constraints defined by the customer, performance and costs.

The aim of the Architectural Design phase is to link the system-level operations to the elements of the structural architecture. This phase starts with the decomposition of the System Blocks into parts, representing the logical elements or components that constitute the entire system. The structure of the system is defined in a BDD and an IBD. Then, the system operations are allocated to the parts. For this operation, if elaboration is needed, the allocation is performed graphically by the realization of a “White box Activity Diagram” of each UC. A “White box Activity Diagram” is a copy of the “Black box Activity Diagram”, where every generic block is divided in “swimlanes”, representing the hierarchical decomposition. In other words, to each “swimlane” is associated a system logical component attaining certain functions.

Finally, the aim of the last phase of the “Design Synthesis” is the definition of ports and interfaces of the product from the system-level to the lower level of architectural decomposition. Starting from the “Use Case White box Activity Diagrams” realized during the Architectural Design, in this phase the “White box Sequence Diagrams” are derived. While the “Black box Sequence Diagrams” show the sequence of operations of the systems, the “White box Sequence Diagrams” focus on the integration of the various subsystems, taking into account the distribution of the operations.

3.3 Step 2: Specify complete and consistent data model and competences

The second step is relative to the collection of the data models, tools and competences available within the design team and necessary for the resolution of the task. Furthermore, the interdependences among the disciplinary models are outlined. Thus, all the relations input/output of each model are extracted, and they will be employed in Step 3 for the formulation of the design problem. The key agent of this step is definitely embodied by the disciplinary expert, who is supported by the architect and the integrator.

As the present dissertation aims at integrating the subsystems design discipline within a collaborative MDO environment, in this Section is described how the on-board systems design is interrelated with the other disciplines. Furthermore, in subsection 3.3.2 an in-house tool developed at Politecnico di Torino for the aircraft subsystems preliminary design is illustrated.

3.3.1 Integration of the on-board systems discipline within a multidisciplinary design context

The on-board systems design discipline is deeply integrated together with all the other disciplines. In Section 1.2 it has been already mentioned that the subsystems mainly affect the aircraft global masses – e.g. Maximum Take-Off Mass (MTOM), Operating Empty Mass (OEM) – volumes, fuel consumption, aerodynamics, RAMS (Reliability, Availability, Maintainability, Safety) specifications, costs, position of the center of gravity. Therefore, the majority of the design choices of the subsystems discipline affects all the other disciplines. In the present subsection, the main design variables relative to the on-board systems discipline that have a significant impact on the other disciplines are identified and discussed. This analysis won’t be quantitative, but qualitative. In other words, the aim of this subsection is to highlight which are the design variables mostly impacting the OAD process, without quantifying the effects of these parameters on all the other design disciplines.

Obviously, the on-board systems design discipline is also affected by the other disciplines, other than by several high level requirements, as number of passengers, speed and altitude along the mission profile. For instance, impacts on the subsystems are due to the aerodynamic hinge moments acting on the mobile surfaces, the aircraft global masses as MTOM and OEM, the fuel flow and the fuel type defined by the propulsion expert. As the following subsection will be devoted to the description of the on-board systems design tool, more details concerning these coupling variables will be provided.

Figure 18 shows the main aircraft design disciplines affected by the on-board systems design variables. This subsection is organized providing per each aircraft subsystem the relative design variables impacting the other disciplines. In conclusion of the present subsection, Table 6 summarizes which design disciplines are affected by the aircraft on-board systems.

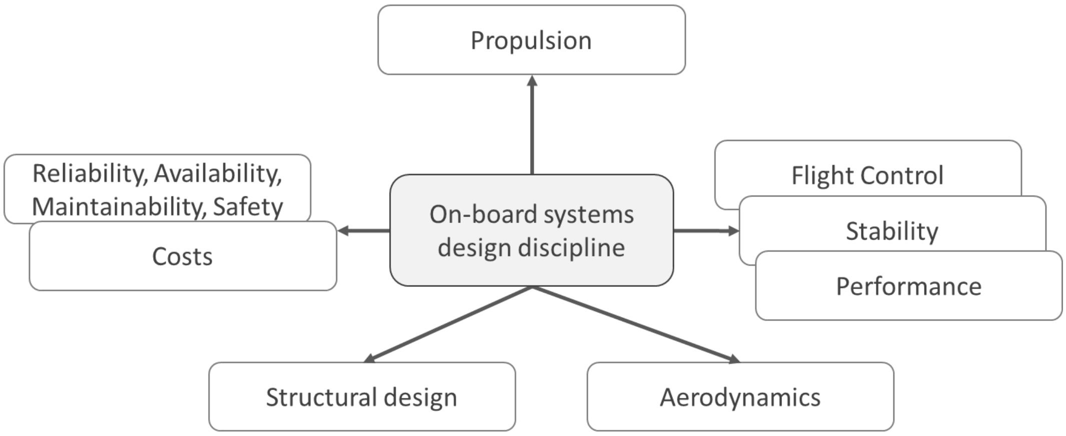


Figure 18: Effects of subsystems on OAD disciplines (adapted from [122]).

Two are the top parameters of the avionics design impacting the other disciplines that are identified. The first parameter encompasses all the functionalities required by the avionic system. From a functional development of this aircraft system, several components are selected, influencing the airplane gross weight and the secondary power extraction [123]. The volume of the avionic equipment might impact the external shape of the fuselage, hence affecting aerodynamics and performance [124]. Also RAMS specifications and costs depend on the installed avionic components, due to their acquisition and maintenance costs. Moreover, the power required by the avionics influences the fuel quantity and cost. Additionally, certain avionic functions – and thus some components – could be required to reduce the structural loads and hence the structural weight [16]. In this work, the functionalities regarding the gust alleviation and the flight envelope protection are considered as example. Other functionalities and components – as the stability and handling qualities control through a Fly-By-Wire (FBW) system and the optimization of the trajectories – affect other design disciplines, as flight control, stability and performance [125]. The second kind of design variable is represented by the required level of redundancy. The installation of new redundant components impacts the aircraft weight and the fuel consumption. Furthermore, the equipment and maintenance costs can be negatively affected by the addition of other avionic components, but the safety level can be improved.

More numerous are the design variables of the FCS affecting the OAD process. The type of actuators (i.e. hydraulic or electric) and the relative supply voltage or pressure determine the dimensions of the components ([27], [126]). Therefore, the weight and the volume of the actuators are influenced by these two parameters. An increment of the actuator weight increases the aircraft empty weight. Furthermore, some actuators might be installed inside fairings, hence their dimensions affect the aerodynamic characteristics [21]. Moreover, the type of power supply has effect on the amount of required secondary power, as different technologies are characterized by different efficiencies. The system architecture, defined by the number of hydraulic or electric lines and by the number of actuators per surface, influences the system mass and hence the OEM. Furthermore, the number of lines and actuators determines the safety level of the system, influencing the RAMS discipline. Finally, the performance characteristics (e.g. actuation speed) of the installed actuators impact the flight handling qualities and the dynamic loads.

Regarding the landing gear, all the design variables relative to the actuation systems (i.e. retraction, steering and braking systems) are similar to those of the FCS. Thus, it is worth presenting other design variables peculiar of this type of subsystem. The most important design choices regard the position and the length of the landing gear struts. For example, the aerodynamics might be heavily affected by the fairings storing the retracted landing gear. Even the shape of the tail cone derives from the installation and dimensions of this kind of system, as ground contacts during the lift-off and rotation manoeuvers in take-off must be avoided [127]. Moreover, the length of the landing gear also influences the distance of the aircraft from the ground and therefore the probability of damage from Foreign Object Debris (FOD). Analogously, the structures should be properly sized for the attachment of the landing gear struts and to store the retracted gears. For instance, the wing kink shall be designed to contain the retracted landing gear inside the wing [128]. It is well known that the main position of the main gear in relation with the aircraft center of gravity influences the flight control during the take-off and the stability on ground. Furthermore, the shift of the aircraft center of gravity during the struts extraction and retraction shall be properly assessed. Another design variable influencing the other design disciplines – namely the structural design and the performance – assessed by the subsystems expert is the number of braked wheels. This parameter affects the ground load due to the deceleration during landing or rejected take-off and it influences the landing distance [127].

Moving to the anti/de-ice system, the main design choice of this subsystem is represented by its technology. As will be explained in the following subsection, multiple options are available for this kind of on-board system. The IPS might be conventional, for instance characterized by a pneumatic (aerothermal or with boots) configuration, or it could be electric. In the first case the hot airflow bled from the engines influences the performance of the propulsion system. Otherwise, in case of the adoption of an innovative system with electric resistors, the efficiency of the propulsion system is again impacted as a percentage of its generated power is extracted and converted in electricity. Also the aerodynamics might be affected by the technology of this subsystem, and thus the flight performance. Aerodynamic drag of the wings can be increased by inflating boots [129]. Regarding the electric IPS, the additional resistive surfaces mounted on the wings might slightly impact the aerodynamic drag, too.

New technologies in the on-board systems discipline regard the ECS, too. As already mentioned in the introductory Chapter, the system architecture impacts the aircraft empty weight and the fuel consumption. Innovative “bleedess” configurations aim at reducing the efficiency losses of the propulsion system, therefore impacting on the needed quantity of mission fuel and on the Direct Operating Costs (DOCs). On the contrary, dedicated air intakes required by this innovative kind of solution entail increments of the aerodynamic drag [130]. Another design parameter assessed by the subsystems expert in accordance with the structural expert is the level of the cabin air pressure. This value impacts the fatigue life of the fuselage, affecting the structural design and the scheduling of the maintenance operations. However, the environment inside the cabin might be made more comfortable thanks to higher pressures [131]. The comfort inside the cabin is also negatively affected by the percentage of air recirculation, as only part of the air results completely fresh and clean [132]. Nevertheless, this percentage improves the efficiency of the propulsion system, hence reducing the extraction of secondary power.

The aircraft stability and flight control is deeply affected by the fuel system. The high fuel weight and the position of the fuel tanks have a great impact on the position of the center of gravity. In this regard, it is worth recalling the odd feature of the Concorde, which transferred a huge percentage of the boarded fuel moving the aircraft center of gravity to compensate shift of the center of pressure during the transition from and to the supersonic flight [133]. However, this feature might be still employed nowadays, in order to reduce the aircraft stability in cruise, hence alleviating the aerodynamic loads acting on the stabilizer [134]. Another design parameter peculiar of the fuel system affecting other disciplines is the volume of the fuel tanks. The dimensions of the tanks indeed affect the thickness of the wing sections, hence impacting of the sizing of the structural elements of the wings and on the aerodynamics, as thicker airfoils entail increment of the aerodynamic drag and vice-versa.

Table 6: Summary of design disciplines affected by on-board systems.

Subsystems	**Design variables**	**Design disciplines**				
		Aerodynamics	**Structural design**	**Flight control, stability and performance**	**Propulsion**	**RAMS, costs**
Avionics	Functionalities selection	•	•	•	•	•
	Number of redundant components		•		•	
Flight Control System	Actuator technology (electric / hydraulic)	•	•		•	•
	Actuator power supply (voltage / pressure)	•	•			
	Number of lines and actuators per surface		•			•
	Actuator speed		•	•		
Landing gear	Strut position	•	•	•		
	Strut length	•	•	•		•
	Number of braked wheels		•	•		
IPS	Ice protection technology (standard, bleedless)	•		•	•	
Environmental Control System	System technology (standard / innovative)	•		•	•	•
	Cabin pressure during cruise		•			•
	Percentage of recirculating air				•	•
Fuel System	Number of tanks and position		•	•		
	Tanks volumes	•	•			
EPGDS/ HPGDS	Supply voltage / pressure		•			•
	Number of lines and generators / pumps		•			•

Finally, the attention is posed on the power generation and distribution systems, namely the EPGDS and the Hydraulic Power Generation and Distribution System (HPGDS). The selection of the main electric voltage or the hydraulic pressure entails effects of the systems weight, hence impacting on the structural design [16]. Given a certain amount of power request, higher values of voltage or pressure bring to weight reductions. Therefore, innovative aircraft are moving towards the adoption of high voltage electric systems and high pressure hydraulic systems. However, the increment of these parameters might impact the safety and the schedule of the maintenance operations, other than affecting costs of the equipment, even if benefits at aircraft-level in terms of reduction of maintenance costs and improvements in reliability are generally envisaged ([96], [135]). Furthermore, the number of distribution lines, as the number of components – namely generators and hydraulic pumps – influence the aircraft empty weight, but also the airplane safety level and the maintenance tasks.

3.3.2 An on-board systems design module

In the last few years, Politecnico di Torino has collected, integrated and developed several subsystems design algorithms and methodologies, which have been coded within an in-house L1 fidelity tool named ASTRID (Aircraft On Board Systems Sizing and Trade-Off Analysis in Initial Design) [20]. ASTRID has been conceived to enable the design of both conventional and innovative subsystems, as More and All Electric architectures and the hybrid propulsion system. In particular, in the context of the current doctoral research, state-of-the-art algorithms for the preliminary estimation of subsystem masses have been modified and adapted for the development of innovative on-board systems. With ASTRID, trade-off studies among different solutions can be performed, eventually selecting the best one, which will be deeply analyzed in successive more detailed design phases. Other than the definition of the global architecture of each on-board system and the preliminary estimation of subsystem masses, the main results of ASTRID include the subsystems shaft power and bleed air off-takes estimated during every segment of the mission profile. Additional outputs interest the component level, in particular with the definition of main equipment with relative specifications as weight, volume and installation. The tool designs both power consuming and power generation systems. The former include the avionics, the FCS, the landing gear, the WIPS, the CIPS, the ECS, the APU system, the furnishing and the fuel system. In the latter category are considered the EPGDS, the HPGDS and the PPGDS. Furthermore, during the doctoral research algorithms for the preliminary design of the hybrid propulsion system have been developed and integrated within ASTRID. The following subsections briefly present the algorithms and methodologies relative to the main subsystems, with particular attention on the hybrid propulsion system.

Flight Control System

The FCS design module allows the preliminary design of the system architecture, with regard to mobile surfaces configuration and actuation system definition. This design module represents a connection between subsystem design and several other specific disciplines, for instance aerodynamics, flight mechanics and handling qualities. Thus, the on-board systems design discipline expects as input the definition of the mobile surfaces, their geometry, positioning, and other requirements, as actuation velocity constraints. A main required input is represented by the aerodynamic loads acting on every control surface. However, ASTRID allows a preliminary estimation – at least concerning primary surfaces as ailerons, elevators and rudders – based on the calculation of the hinge moments [136].

From the loads estimation results and the requirements of actuation velocity, the power of each control surface is evaluated, therefore assessing the power budget during the whole mission profile. This output is required for the sizing of the actuation system, in terms of technology, number of actuators and redundancies, their dimensions, stall and nominal forces and velocity and their installation. Different types of actuation systems are considered within ASTRID, in accordance with the current tendency but also short-term future trends. Thus, not only conventional linear and rotary actuators are sized, but also electrically powered systems as EHAs and EMAs. Therefore, statistical ratios power/mass [kW/kg] or force/mass [N/kg] are employed to determine the effective mass of the components. Similar ratios are gathered from databases and brochures, thus obtaining the volume required by the equipment. Additionally, ASTRID contains methodologies for the mass estimation of the whole on-board system, considering both conventional and innovative technologies. In the first case, semi-empirical formulations developed by Torenbeek [68], Roskam [69] and Raymer [70] are used. These equations are valid in case of conventional actuation systems. However, new algorithms are necessary for the weight estimation of innovative subsystems. The FCS weight can be given by the sum two parts, the former regarding the surface actuators and the latter encompassing all the kinematics. In this work, it is assumed that the weight of the kinematics of the FCS results unchanged depending on the technology level (i.e. conventional or More/All electric). Therefore, the mass of innovative actuation systems can be calculated by means of the same semi-empirical formulations used for the conventional architectures, in which conventional actuators – characterized by well-known power/mass ratios – are considered. Assuming different power/mass ratios of the new actuators, also innovative architectures might be evaluated.

Landing gear

Three actuation subsystems are sized within the landing gear design module: the retraction, the steering and the braking subsystems. Again, this module estimates the hydraulic or electric power required by the landing gear actuation systems all along the mission profile and the weight of the entire subsystem and the main components, as structures, wheels and actuators.

The retraction system power is evaluated considering only the gravitational forces acting on the landing gear. In this preliminary phase of the design process, the other forces as aerodynamic drag, friction and dynamic load can be neglected, as stated in [137]. Thus, the maximum mechanical power (P_{max} [W]) required during the retraction of each single landing gear strut is given by:

$$P_{max} = m_{LG} \cdot g \cdot b \cdot \left(\frac{\partial\vartheta}{\partial t}\right)_{max} \qquad \text{eq. 1}$$

where the parameters of eq. 1 are (see Figure 19 as reference):

- m_{LG} [kg] is the mass of the landing gear strut;
- g [m/s^2] is the gravitational acceleration;
- b [m] is the distance between the leg hinge and the strut center of gravity, which is supposed to be coincident with the center of the wheel;
- $\left(\frac{\partial\vartheta}{\partial t}\right)_{max}$ [rad/s] is the maximum angular speed of the landing gear leg during the retraction.

From the results obtained by eq. 1, the hydraulic or electric power required by the retraction system is evaluated considering different efficiency values of the actuators.

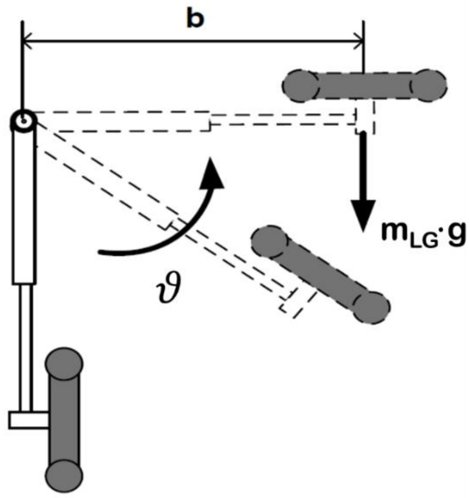


Figure 19: Landing gear retraction schema (adapted from [20]).

Concerning the methodology for the power estimation of the steering subsystem, firstly the static load L_N [N] acting on the nose landing gear is evaluated by means of eq. 2.

$$L_N = \frac{m}{l} \cdot W \qquad \text{eq. 2}$$

The distances l [m] and m [m] respectively between the landing gear legs and the main landing gear with the aircraft center of gravity are clearly shown in Figure 20. W [N] instead refers to the aircraft maximum weight during taxi.

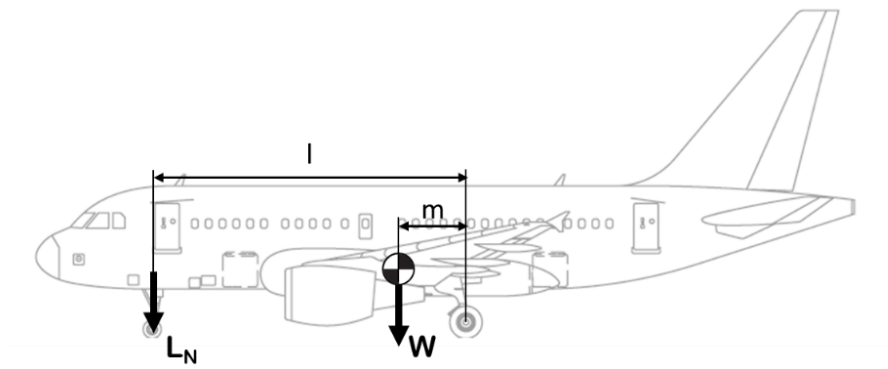


Figure 20: Estimation of the gravitational load acting on the nose landing gear.

Given the nose landing gear load, the steering moment M_S [Nm] is evaluated (see eq. 3), considering also the sliding friction f between the wheels and the taxiway. The parameter $a = \sqrt{(R^2 - {R_M}^2)}$ is represented in Figure 21 [127], and is function of the difference between the radius R of the wheel and the radius R_M [m] of the tire squeezed by the gravitational force acting on it.

$$M_S = f \cdot L_N \cdot a \qquad \text{eq. 3}$$

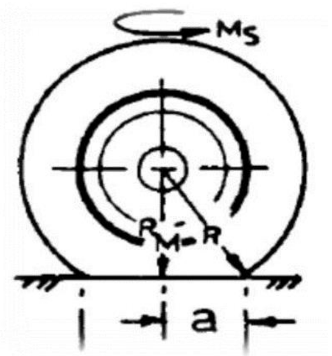


Figure 21: Estimation of the parameter “a” of the pressed tire [127].

Successively, the steering actuation power is derived multiplying the resulting moment by the steering angular speed evaluated at the leg axis and by the actuator efficiency.

Finally, the braking system sizing procedure is hereafter presented. It derives from the methodology proposed by Currey [127]. The design process estimates the power required by the subsystem to completely stop the aircraft in two conditions. The former refers to the event of a Rejected Take Off (RTO), while the latter is considered during the landing phase. Firstly, the force required to stop the plane is evaluated. The aircraft is supposed to have a mass equal to the MTOM or to the Maximum Landing Mass (MLM), according to the condition. Then, the braking distance is set. In case of RTO, this value can be estimated from a hypothesized value of the Balanced Field Length (BFL) distance, which is function of the TOFL requirement. During the evaluation of the landing condition, the braking distance is equal to the Landing Field Length (LFL), which generally is a TLAR. The MTOM or LFL, together with the aircraft V_1[3] or approach speed, entails the calculation of the airplane deceleration on the runway. This value multiplied by the MTOM or the MLM returns the required braking force, which multiplied by the wheel radius gives the braking torque T_{br} [Nm]. Then, the force F_{br} [N] exerted by the brakes is evaluated by means of eq. 4:

$$F_{br} = \frac{F_{br} \cdot \%_{br}}{N_{wh} \cdot N_d \cdot R_f \cdot f_{br}} \qquad \text{eq. 4}$$

where:

- $\%_{br}$ represents the percentage of braking force due to the braking system. Other means are employed to decelerate the airplane, as aerodynamic drag and thrust reversers;
- N_{wh} is the number of braked wheels;
- N_d is the number of disks per wheel;
- R_f [m] is the wheel radius where the resultant braking force is supposed to be exerted. This radius can be assumed equal to 2/3 of the wheel radius;
- f_{br} is the sliding friction among the wheel disks.

The mechanical power required by the braking system is therefore calculated multiplying F_{br} by the distance run by each single brake piston and dividing this result by the duration required by the piston to complete a stroke. At this point, the methodology ends with the evaluation of the actuation system power. As electric brakes are already installed on some aircraft (for instance on the Boeing 787 [138]), both hydraulic and electric power can be computed, considering different efficiencies.

Concerning the weight of the landing gear, several methodologies are present in literature, for instance [68], [69], [70] and [127]. These methods are effective in case of conventional systems, as hydraulic powered systems. However, in case of

[3] The V_1 is the aircraft speed during the take-off manoeuver at which the Accelerate-Stop Distance Required (ASDR) is equal to the Take Off Distance Required (TODR), on the basis of the airplane mass, engine thrust, aircraft configuration (e.g. flaps extracted) and runway condition. In case of a critical engine failure during take-off, the airplane must abort the procedure whether its speed is lower than V_1, otherwise the manoeuver shall be completed.

innovative actuation systems, the methodologies can be corrected considering a different power-to-weight ratio of the actuators, knowing that typically their weight ranges between 9% and 21% of the total landing gear weight [127], according to the type of airplane.

Ice Protection System

Analogously to the other aircraft on-board systems, ASTRID allows the sizing of conventional and innovative anti-ice and de-ice systems. Therefore, the hereafter described methodology is used to compute:

- hot airflow bled from the engines, required to heat the wings leading edge and engine cowls (Aerothermal IPS);
- pressurized air tapped from the engines, inflated inside pneumatic boots installed inside the wings leading edge (Pneumatic IPS);
- electric power required to protect the wings and other small parts, such as antennas, propellers, sensors (Electric IPS).

The proposed design methodology starts with the definition and the evaluation of the area S_{ice} [m^2] of all the surfaces to be protected. This value should be estimated by a different disciplinary expert, as an aerodynamics or a flight mechanics expert. Alternatively, in the case of the wings leading edge, Liscouët-Hanke [21] describes how to assess the span-wise length of the protected area. This value together with a chord-wise length supposed in a first approximation returns the area of the surface that should be de-iced.

Once all the protected surfaces have been assessed, the described methodology proceeds with the sizing of the three typologies of IPS.

<u>Aerothermal IPS</u>

The aerothermal system sizing methodology coded within ASTRID is treated in [139] in greater details. Briefly, it entails the estimation of the airflow $\dot{m}_{ice}$ [kg/s] required by the system by means of eq. 5:

$$\dot{m}_{ice} = \frac{\dot{q}_{tot} \cdot S_{ice}}{c_p \cdot \left(T_{pn} - T_{ice}\right)} \qquad \text{eq. 5}$$

From experimental data, the heat flow per unit area $\dot{q}_{tot}$ [kW/m^2] can be assumed equal to 10 kW/m^2 [139]. S_{ice} [m^2] represents the area of the surface to be protected, while c_p=1.005 kJ/(kg·K) is the air specific heat at constant pressure. The parameters T_{pn} (K) and T_{ice} (K) are respectively the air temperature of the pneumatic system and the ice temperature.

Pneumatic IPS

Again, the proposed methodology estimates the airflow required by the pneumatic boot used to break the ice layer accreting on the wings surface. Given the air density inside the boot ρ_{boot} [kg/m^3], the boot volume V_{boot} [m^3] and the inflation time t_{boot} [s], the airflow $\dot{m}_{boot}$ [kg/s] is calculated by means of eq. 6:

$$\dot{m}_{boot} = \rho_{boot} \cdot \frac{V_{boot}}{t_{boot}} \qquad \text{eq. 6}$$

Electric IPS

An innovative type of IPS is represented by the electric anti-ice and de-ice system. The surface is thus protected by the heat generated by a layer of electrical resistors located on the wings leading edge. The electric resistors are placed as schematically depicted in Figure 22, distinguishing two types of area. The first type of area is constantly de-iced, while the latter type is cyclically warmed up. As described in [25], these two zones are characterized by a different heat flow per unit area. The AIR 1168/4 [140] by SAE International suggests the values of heat flow per unit area equal to 18.6 kW/m^2 for the continuously protected area and 34.1 kW/m^2 for the cyclically de-iced area. Using the Boeing 787 as reference, Meier *et al.* [25] instead computes lower values, respectively equal to 11.82 kW/m^2 and 27.25 kW/m^2. As newer electric IPSs might be characterized by a reduction of the electric power demand, similar values computed by Meier *et al.* are adopted in ASTRID. In particular, for the continuously protected zones the heat flow per unit area is set to 12 kW/m^2, while a value equal to 28 kW/m^2 is chosen for the cyclically de-iced area. Thus, the proposed methodology entails the computation of the optimized number of the two kinds of area, consequently estimating the total required electric power. The implemented optimization routine estimates the number of the two types of area, with the objective of minimizing the electrical power consumption, under the constraint of effectively de-icing the surface avoiding any ice accretion. Indeed, a high number of cyclically protected zones would entail a low de-icing duty cycle per area, then dramatically bringing to ice formation.

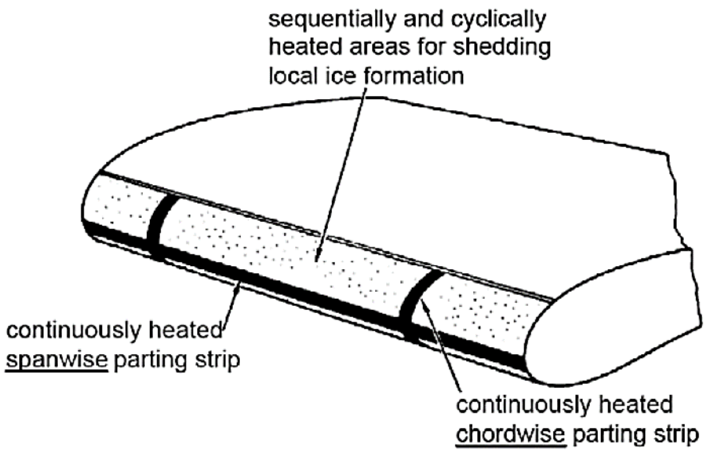


Figure 22: Continuously and cyclically heated areas on the wing surface [25].

The design of the IPS ends with the preliminary mass estimation. Methodologies (e.g. [68], [69] and [70]) in literature have been developed with this aim. However, all these methods are based on old statistical basis. Thus, they are rather reliable for conventional systems, while new methodologies are required for the electric IPS. Knowing the mass of the conductive film per unit length of the extension of the protected area, the weight of this innovative subsystem can be easily calculated. However, on the basis of the studies proposed in [97], to a first approximation the weight of the electric IPS can be assumed equal to 60% of the weight of an analogous aerothermal system.

Environmental Control System

Aim of this module is the sizing of the aircraft air conditioning system. Two are the main results of the design. The first one is the evaluation of the airflow required by the system to maintain the air temperature in cabin within a predetermined range. In a conventional architecture, the airflow is tapped from the jet engine compressors, then expanded and finally cooled down by two or more Cold Air Units (CAUs). In this case, the estimated airflow is used to size the pneumatic system, as will be described later. Otherwise, innovative solutions are characterized by the "bleedless" configuration, as introduced in subsection 1.2.1. Therefore, the proposed methodology estimates the electrical power required to move dedicated compressors pressurizing external air gathered from scoops air intakes. The second main result is a preliminary evaluation of the ECS mass, both for conventional and innovative systems.

The sizing procedure is outlined by several steps. The first step is the estimation of the total maximum and minimum thermal loads inside the cabin. The total thermal loads are derived by the sum of four partial thermal loads:

1) Heat flow through the fuselage, which depends on the type of materials that constitute the fuselage wall (i.e. structure, insulation layer) and the difference of temperature between the airflow and the aircraft external surface;
2) Solar heating, i.e. the heat flow of the sun through windows;
3) People physiological heating, i.e. heat generated by passengers and flight crew;
4) Cabin equipment heating, which mostly considers the heat generated by the avionics or other equipment, for instance lights, In-Flight Entertainment (IFE) system.

Two opposite conditions are considered in the evaluation of the thermal loads. The first one refers to the request of maximum heating. In this case, a cold night flight without passengers aboard is used for the evaluation. The second condition regards the requirement of maximum cooling. In the calculation of the thermal load of this case, the aircraft is supposed to be parked on ground, during a warm sunny day, with the maximum number of passengers aboard.

Next, the design methodology proceeds with the evaluation of the airflow $\dot{m}_{ECS}$ required to nullify the thermal load w_{TOT} [kW] ensuring a set internal environment air temperature T_{cab} [K] and a maximum temperature of the airflow entering inside the cabin T_{ic} [K]:

$$\dot{m}_{ECS} = \frac{w_{TOT}}{c_p \cdot (T_{ic} - T_{cab})} \qquad \text{eq. 7}$$

The obtained airflow is used to size all the main system components, as the CAUs. In ASTRID are sized both ACMs and Vapor Cycle Machines (VCMs). The former category of CAU employs the air itself to reduce the airflow temperature, while the latter exploits the change of state of a refrigerant such as Freon®. Hence, the design methodology evaluates the airflow $\dot{m}_{PN}$ [kg/s] required by the CAU from the pneumatic system or from dedicated compressors of the innovative “bleedless” concept:

$$\dot{m}_{PN} = \dot{m}_{ECS} \cdot (100 - \%_{rec}) \quad \text{eq. 8}$$

where the percentage of recirculation $\%_{rec}$ could be set in the range 0%-50%, according to the level of technology of the CAU. Current ACMs reduce the airflow temperature down to -20°C. Therefore, the air entering in the cabin should be mixed with warmer airflow.

According to the regulation [141], the result of eq. 8 shall guarantee at least 0.00415 kg/s of clean air per person on board. Thus, in case the resulting airflow is not compliant with the regulation, it must be immediately corrected.

In the case of maximum heating condition, part of $\dot{m}_{PN}$ should by-pass the CAU. The percentage of by-pass is a function of the target air temperature in cabin and of airflow temperature exiting from the conditioning machine.

In the case of a “bleedless” architecture, the electrical power $P_{ECS_{compr}}$ [kW] required by the dedicated compressors is calculated:

$$P_{ECS_{compr}} = \frac{\dot{m}_{PN} \cdot c_p \cdot \left(T_{fc} - T_{ext}\right)}{\eta_{compr} \cdot \eta_{elect}} \quad \text{eq. 9}$$

The air temperature of final compression T_{fc} [K] might be easily calculated knowing the compression ratio of the dedicated compressor; T_{ext} [K] is the external air temperature; η_{compr} is the compression efficiency while η_{elect} is related to the conversion losses of the electric motor moving the compressor.

The preliminary estimation of the subsystem mass terminates the design of the ECS. Again, the methods [68], [69] and [70] are useful to evaluate the mass of the conventional system. The methodologies can be adapted for the assessment of innovative architectures: given the power-to-mass ratios of dedicated compressors and electric motors, the mass of the “electric” air conditioning system is indeed determined.

Fuel System

The objective of the following design module is the preliminary assessment of the power and mass budgets of the fuel system.

Concerning the power budget, the only components whose the electric supply power is estimated in ASTRID are the fuel pumps. The electric power required by the pumps depends on the fuel flow generated and by the supply pressure. In this regard, two are the main functionalities of the fuel pumps: to deliver fuel to the engines or the APU and to transfer the fuel from one tank to another. Therefore, the fuel flow depends on the engines demand, while the pressure is a function of:

- Required fuel pressure at destination. This parameter is equal to the fuel pressure inside tanks in case of fuel transfer or it varies within a certain range – generally 1-3 bar – as prescribed by the propulsion expert.
- Pressure drops along the conduit from the tank to the destination, due to the length, material, and route of tubing, and the difference in height.

The fuel system can be considered almost unvaried since the beginning of the aviation. Minimal innovations have been made concerning the materials of the components and the technology of the fuel pumps. Therefore, the methodologies present in literature (as [68], [69] and [70]) are valid for a preliminary estimation of the system mass.

Pneumatic Power Generation and Distribution System

Once all the power demanding system have been sized, the design methodology proceeds with the dimensioning of the power generation and distribution systems.

The dimensioning method of the PPGDS sums the airflows required by the de-ice and air conditioning systems. Other values of airflow can also be considered, as taking into account the pressurization of the fuel tanks or the presence of additional pneumatic turbines.

Concerning the system mass, the most well-known methodologies allocate the masses of the main system components within the ECS or the traditional IPS. According to the ATA 100 chapters, the PPGDS (ATA chapter 36) is constituted of distribution and indicating equipment. In the preliminary mass estimation of both conventional and innovative subsystems it is legitimate to neglect the weight of the indicating components, while including the distribution part inside the ECS and the IPS, if this system is aerothermal.

Hydraulic Power Generation and Distribution System

The design of the HPGDS entails the sizing of the main hydraulic components on the basis of the results obtained by the analysis of the FCS and landing gear. In particular, ASTRID is focused on the preliminary sizing of the hydraulic pumps.

The estimated power per flight segment of the mission profile is employed for the preliminary definition of the hydraulic system architecture (i.e. number of hydraulic circuits, reservoirs, types of pumps, accumulators). The procedure starts with the allocation of all the hydraulic users (e.g. each single actuator) to each circuit. Each circuit is supplied by a hydraulic pump. Thus, the allocation of the users entails the sizing of the hydraulic pumps. Given the required total of hydraulic power P_{hydr} [kW], and fixed the value of system pressure, the hydraulic oil flow $\dot{m}_{hydr}$ [l/s] is calculated:

$$\dot{m}_{hydr} = \frac{P_{hydr}}{p_{hyd\ sys}} \qquad \text{eq. 10}$$

In ASTRID two types of hydraulic pumps are considered. The first one is mechanically moved by the engines by means of the AGB. In this case, the oil flow generated by the pump depends on the angular speed of the high pressure engine shaft. Therefore, the hydraulic pump is typically sized for the descent phase condition, in which the engine is set in idle (low angular speed) but the hydraulic power demand might be high. Hence, the hydraulic pumps might result oversized for the other mission segments. In order to resolve this issue, a second type of pump is being developed. These pumps are driven by electric motors, entailing a more efficient power supply. The HPGDS design module therefore entails the estimation of the mechanical or electric power demanded by the pumps.

Concerning the subsystem mass, an estimation method is provided by Roskam [69]. This methodology is suitable for traditional 3000 psi (about 20.7 MPa) pressure hydraulic systems. The mass of newer 5000 psi (about 34.5 MPa) pressure systems can be evaluated reducing the conventional system mass by 28.3% [126].

Electric Power Generation and Distribution System

The last power generation system design module concerns the EPGDS. The methodologies and algorithms implemented in ASTRID are in line with the evolution of the electric standards and the introduction of novel electric components. Therefore, other than traditional electrical voltages – i.e. 28 V DC and 115 V AC (400 Hz) – new standards are considered, as 270 V DC, 235 V AC variable frequency and 115 V AC variable frequency. The design module entails the sizing of traditional and new electrical machines, i.e. generators and power converters. Furthermore, a preliminary system mass estimation is assessed.

The design process starts with the definition of the total number of generators. Generally, the electric system architecture is characterized by one generator per engine and one generator per APU. The generator voltage is defined on the basis of the power budgets of all the power demanding systems previously designed. As more than one kind of electric voltage is generally needed, power converters shall be designed to supply all the users installed on board. Then, the results of required electrical power and voltage of each user active during every segment of the mission profile are employed for the dimensioning in terms of maximum power of all the electric generators and power converters.

Once the maximum electric power of the main components is assessed, the mass of the electrical machines can be obtained on the basis of the power-to-mass ratios provided by the manufacturers. In this way, the mass of both conventional and innovative components can be evaluated. However, for the estimation of the entire system (including cables, controls and batteries) can be evaluated by means of conventional methodologies (e.g. [68], [69] and [70]). In this regards, it is worth reminding that the increment of the electric system voltages brings to a reduction – considering unvaried the demanded electric power, the cable lengths l_{cond} [m] and the adopted cable materials (same density and same electric resistivity ρ_{electr} [Ω·m]) – of the mass of conductors. The evolvement of Ohm's law brings to eq. 11, where the volume Vol_{cond} [m^3] of the conductor is pointed out:

$$Vol_{cond} = \rho_{electr} \cdot l_{cond}^2 \cdot \frac{i}{V} \qquad \text{eq. 11}$$

If the electric power demand remains constant, the variation of the electric current i is inversely proportional to the variation of the electric voltage V. Therefore, according to eq. 11, moving for instance from 115 V to 270 V would bring to a reduction of Vol_{cond} of about 80%, which entails the same reduction of the mass of the conductor (hypothesis of same density). This calculation can be done to obtain a first approximate assessment of the wiring mass on the basis of the system voltages. However, it shall be noted that other considerations should be included in more detailed evaluations, as the possible variation of the dimensions (and hence masses) of the wires insulation due to the increment of the electric current. Furthermore, it is worth noting that the dimensions (in particular the section) of the electric cables in aeronautics are characterized by a minimum physical limit [142].

Hybrid-Electric Propulsion System

The methodology described in the present subsection aimed at the preliminary design of hybrid propulsion systems is composed by four parts [143]. The first part consists in the quantification of the electric power and electric energy required during the taxi phases. In the second part of the methodology, the same parameters are evaluated for the take-off phase, when the total propulsive power is generated by both the electric motor and the ICE. The third part is aimed at assessing the minimum safety altitude. This altitude is the minimum one at which the aircraft could safely land at the same departure runway in case of a thermal engine failure during the climb soon after the take-off maneuver. At the end, the methodology allows the sizing of the components installed within the HEPS. In particular, the masses of the electric motor, the batteries and the mechanical transmission are assessed, thus obtaining the total hybrid propulsion system mass.

The first step of the proposed methodology is devoted to the determination of the electric power and electric energy required during taxi. It is worth noting that two taxi phases are included within a typical mission profile. The algorithms hereafter proposed can be indeed employed for both the taxi out and the taxi in phases. However, it should be recalled that before taking off the ICE shall have reached a minimum temperature in the cylinder head. Therefore, an interval of a few minutes before the take-off clearance would be required to warm-up the thermal engine. This would nullify some possible benefits of the hybrid-electric propulsion system, as the reduction of fuel consumption and pollution, though the aircraft would reduce the amount of emissions nearby populated areas. During the taxi in phase instead the advantages of the electric propulsion would be effective.

The necessary propeller shaft power $P_{prop\,shaft}$ [W] required by the aircraft during taxi is estimated by eq. 12:

$$P_{prop\,shaft} = \frac{m \cdot g \cdot \mu \cdot v}{\eta_{prop_{taxi}}} \qquad \text{eq. 12}$$

in which m [kg] represents the aircraft mass, g=9.81 m/s^2 is the gravitational acceleration, v is the airplane speed during taxi and $\eta_{prop_{taxi}}$ is the propeller efficiency. The rolling friction coefficient μ might assume values ranging between 0.05 and 0.3 [104], according to the taxiway surface, as collected in Table 7:

Table 7: Rolling friction coefficient μ in different taxiway surfaces [104].

Taxiway surface	μ
Concrete	0.02 – 0.03
Asphalt	0.02 – 0.03
Hard Turf	0.05
Short grass	0.05
Long grass	0.10
Soft ground	0.10 – 0.30

Dividing the value of propeller shaft power obtained in eq. 12 by the electric motor efficiency (η_{EM}), the required electric power $P_{electr_{taxi}}$ [W] is calculated:

$$P_{electr_{taxi}} = \frac{P_{prop\ shaft}}{\eta_{EM}} \qquad \text{eq. 13}$$

Then, the electric energy E_{taxi} [J] is estimated through eq. 14, where the power required by the electric users (P_{users} [W]) is also taken into account.

$$E_{taxi} = \frac{(P_{electr_{taxi}} + P_{users}) \cdot t_{taxi}}{\eta_{batt_{taxi}}} \qquad \text{eq. 14}$$

The energy resulting from eq. 14 is function of both the battery discharge coefficient ($\eta_{batt_{taxi}}$) and the duration of the taxi phase (t_{taxi} [s]). This duration derives from the distance travelled during this phase and the taxi speed.

In the second part of the proposed methodology, the surplus of propulsive power required during take-off is evaluated. This mechanical power $P_{electr_{TO}}$ [W] is estimated by means of eq. 15:

$$P_{electr_{TO}} = \frac{\frac{T_{TO} \cdot v_{TO}}{\eta_{prop_{TO}}} - P_{thermal\ engine}}{\eta_{EM}} \qquad \text{eq. 15}$$

where:

- T_{TO} [N] is the propeller traction required during take-off.
- v_{TO} [m/s] is the aircraft maximum speed during the take-off phase.
- $\eta_{prop_{TO}}$ is the propeller efficiency in take-off.
- $P_{thermal\ engine}$ [W] is the maximum power of the selected ICE. This value must be lower than the total propulsive power required during take-off.

From the value of the required surplus of propulsive power, the electric energy is estimated employing the eq. 14, in which $P_{electr_{TO}}$ is considered instead of $P_{electr_{taxi}}$, and the duration of the phase is relative to the take-off. Moreover, during this phase, the power of the electric users is taken into account.

Then, the methodology allows the evaluation of the minimum altitude at which the aircraft could safely land in case of a failure of the piston engine. The diagram depicted in Figure 23 shows this type of emergency.

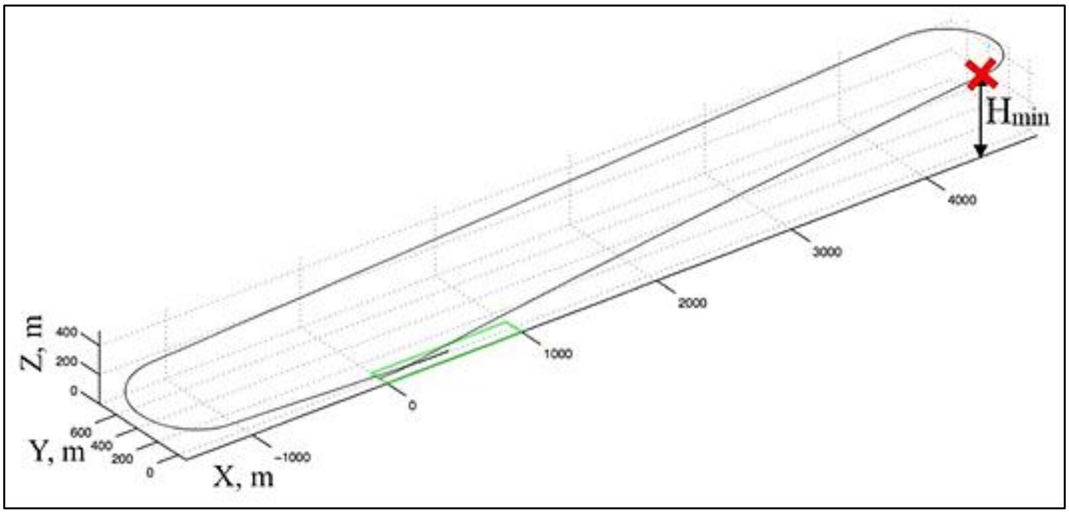


Figure 23: Schema of the emergency descent after take-off [144].

The airplane is supposed to have a malfunctioning affecting the thermal engine at a certain altitude denoted with H_{min} (the red "X" sign in the diagram). After the failure, the aircraft should attempt a final approach and an emergency landing performing a first turn (crosswind leg), traveling the straight line parallel to the runway (downwind leg) and finally turning again (base leg). This procedure is different from the teardrop maneuver proposed by Rogers [145]. In this case, after the failure the aircraft performs a turn back to the runway, towards the opposite direction of the take-off. However, the Federal Aviation Administration (FAA) discourages this procedure, due to the catastrophic event of a potential frontal impact against another aircraft. This notwithstanding, some results of Rogers' study are considered in this thesis. In particular, the bank angle during the turns is taken as reference.

During the entire emergency procedure, the aircraft is powered by the electric motor. The propulsive power generated by the electrical machine ensures a minimal reduction of the descent rate, though it is not sufficient to guarantee a rectilinear flight. However, it allows an enhancement of the safety level decreasing the minimum safety altitude.

The following equations allow a first simplified estimate of H_{min} [m] and the time $t_{emergency}$ [s] required to perform the entire emergency operation:

$$H_{min} = \frac{\Delta H_{turn_1} + L_{TO} \cdot \tan\gamma_{rett} + \Delta H_{turn_2}}{1 - \frac{\tan\gamma_{rett}}{\tan\gamma_{climb}}} \quad \text{eq. 16}$$

$$t_{emergency} = \frac{\Delta H_{turn_1} + \Delta H_{turn_2}}{v_{turn} \cdot \sin\gamma_{turn}} + \frac{H_{min} - (\Delta H_{turn_1} + \Delta H_{turn_2})}{v_{rett} \cdot \sin\gamma_{rett}} \quad \text{eq. 17}$$

where ΔH_{turn_1}[m] and ΔH_{turn_2} [m] are the altitude steps lost during the crosswind and base legs respectively, γ_{rett} [rad] and γ_{climb} [rad] represent the descent and climb angle in the rectilinear path and before the failure event, γ_{turn} [rad] is the descent angle during the two turns and L_{TO} [m] is the take-off distance. Moreover, it can be included in H_{min} an additional altitude step of at least 15 m, in order to guarantee a stabilized approach.

Simple flight mechanics equations might be employed to easily calculate the values of the three altitude steps. Assuming a bank angle $\varphi = 45°$ as recommended in [145], the altitude lost during the first turn is equal to:

$$\Delta H_{turn_1} = \pi \cdot \frac{{v_{turn}}^2}{g \cdot \sqrt{\frac{1}{\cos^2\varphi} - 1}} \cdot \tan\gamma_{turn} \quad \text{eq. 18}$$

The eq. 18 can be utilized to estimate the altitude lost during the second turn, too.

The amplitudes of the two angles γ_{turn} and γ_{rett} depend on the aircraft speed during the procedure. Therefore, as suggested by Rogers [145], the turns might be performed at $v_{turn} = 1.05 \cdot v_{stall\,(clean\,config.)}$. However, in compliance with the CS-23 regulation, the aircraft is here supposed to perform the two turns at $v_{turn} = 1.2 \cdot v_{stall\,(TO\,config.)}$. The stall speed during the turns shall indeed include the load factor due to the bank angle. The rectilinear path is flown with the speed v_{rett} corresponding to the attitude of the maximum L/D ratio, thus maximizing the gliding distance. Moreover, the descent angles are affected by the propulsive power P_a [W] generated by electric motor and by the power P_n [W] required to counteract the aerodynamic drag. The descent angle γ [rad] during the turns and the downwind leg are given by the following general formulation:

$$\gamma = \frac{P_a - P_n}{v_{emergency} \cdot m \cdot g} \qquad \text{eq. 19}$$

where P_n and the airplane speed $v_{emergency}$ [m/s] are relative to the turns or the rectilinear segments, while the propulsive power P_a [W] is the maximum power required from the electric motor during the take-off.

As done for the taxi and take-off phases, the energy stored in the battery for this non-nominal situation is finally evaluated.

The mass estimation of the main equipment of the hybrid propulsion system concludes the proposed design methodology. The mass of the electric motor is obtained dividing the maximum required power at take-off by the power-to-mass ratio given by the manufacturer. In particular, different types of electric machines might be installed within hybrid propulsion systems, as DC motors, Induction Motors (IMs), Permanent Magnet (PM) brushless motors and Switched Reluctance Motors (SRMs). In this regard, a comparative study among these types of electric machines can be found in [146], where characteristics, advantages and disadvantages are outlined. Although the study concerns ground Hybrid Electric Vehicles (HEVs), the outcomes of the analysis conducted by Zeraoulia *et al.* [146] can be supposed valid for the aeronautical case, too. The batteries mass is calculated instead dividing the energy required during taxi, take-off and emergency, by the energy-to-mass ratio gathered from statistical databases or datasheets (see Table 8 as an example). Concerning the mechanical transmission, deep and detailed studies could be conducted for the mass estimation of the gears, analyzing the type of materials and the relative density property. However, in this work the mass of the mechanics $m_{mechanics}$ [kg] is preliminarily assumed proportional to the power P_{ICE} $[hp]$ of the ICE:

$$m_{mechanics} = 0.04 \cdot P_{ICE} \quad \text{eq. 20}$$

Table 8: Specific energy ranges of some batteries [147].

Battery	Specific energy [Wh/kg]
Ni-Cd	40-60
Ni-Zn	60-65
Ni-MH[4]	60-70
Li-Ion	90-130
Li- Polymer	155

In conclusion, it is worth noting that several are the impacts of the design of the HEPS on the OAD process. The main effect concerns the variation of the OEM, due to two aspects. From one side, the downsize of the ICE could bring to a lighter propulsion system. However, the introduction of new components as the batteries, the mechanical transmission, and a larger electric machine, would cause a weight penalty. Additionally, the downsize of the thermal engine has a relevant impact on the Specific Fuel Consumption (SFC), as the ICE is generally optimized for the cruise condition. The installation of a less-powerful thermal engine sized for the cruise phase would imply a reduction of consumed fuel in flight, hence entailing a reduction in polluting emissions and operating costs. Real values of the SFC could be specified by engines manufacturers. Nevertheless, the smaller thermal engine might entail a degradation of the performance, as the increment of the climb phase duration or the reduction of the ceiling.

Other effects are significant, although not studied in the context of the present doctoral research, as concerning aerodynamics, RAMS specifications, volumes, installation aspects.

[4] Ni-MH: nickel metal hydride battery

3.4 Step 3: Formulate design problem and solution strategy

Once all the design competences required to design the new product have been collected, an MDO architecture is formulated by the architect to solve the MDO problem. The formulation of the design problem is the main aim of the third step of the development process. This step assumes great importance due to the fact that MDO problems are growing in size and complexity. This happens because of the high number of the coupling variables resulting from the disciplines involved within the design process. Therefore, the MDO problem should be organized to obtain the desired solution minimizing costs, time and effort. In other words, an MDO architecture should be formulated defining how the disciplinary analysis models are coupled together and how the optimization problem shall be addressed [4]. Formulating an MDO architecture means to organize the disciplinary competences – e.g. ordering their sequence during the analyses, defining the data flow – and determining the design variables, the objective functions and the constraints. Due to the high complexity of an MDO architecture, a visual description might be helpful to comprehend the MDO problem. An example of visual description is given by the eXtended Design Structure Matrix (XDSM), a graph of which an example is depicted in Figure 24 [4]. The XDSM is proposed by Martins *et al.* [4] to enhance the visualization of the interconnection among the modules integrated within a MDO process.

Examples of this type of diagram will be described in Section 4.3. In particular, more than one XDSM will be presented, according to the kind of design problem.

The XDSM depicts the connections among the design disciplines (or disciplinary tools), showing both the data and process flows. The data flow is represented by the grey lines. Each vertical line connected to a discipline model represents inputs, while horizontal lines denote outputs. The *design variables* x_i represent the parameters under the control of an optimizer. In other word, an optimization code may vary the values of the design variables within a predetermined range to comply with the *objective functions*, for instance minimizing costs or masses. The optimized results are denoted as y_i^*, corresponding to the input variables x_i^* of the design space. The outputs of each design discipline are represented by the data flowing on the horizontal lines. These parameters are denoted as *state variables*. If these values are exchanged with other disciplines, they are referred as *coupling variables* y_i.

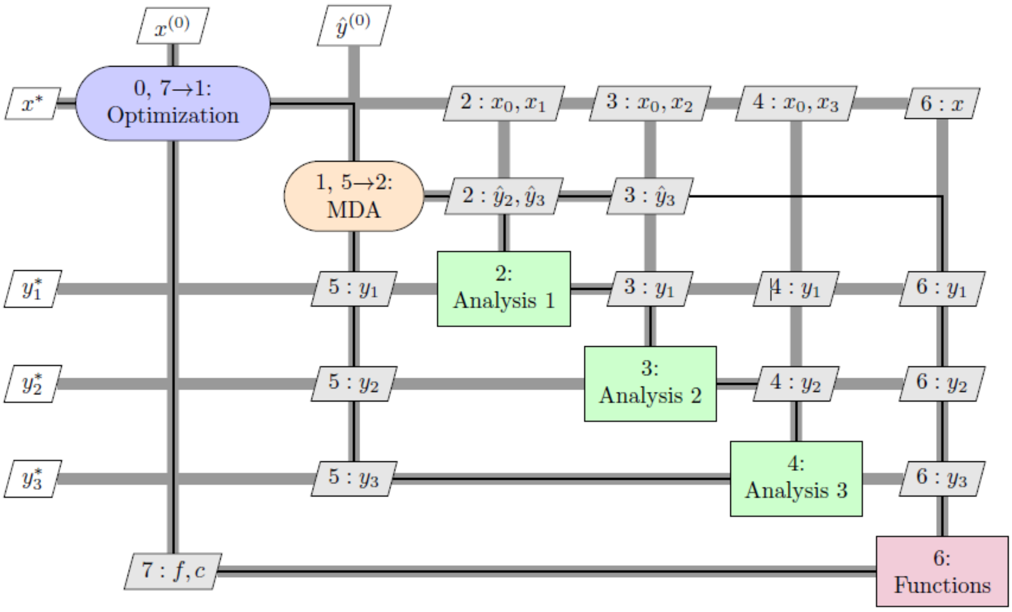


Figure 24: Example of XDSM [4].

Several design problems might be assessed. Un-converged Multidisciplinary Design Analysis (MDA) processes entail the assessment of the response of a set of disciplinary modules given a certain ensemble of design variables. Iterations and backward loops would be necessary to converge to the design solution (converged MDA). Differently from the MDO problem, MDA processes don't entail the determination of an optimized solution. Single and multi-objective MDO problems indeed might be setup to obtain an optimized solution. In this regard, a multi-objective MDO problem based on the Fuzzy Logic will be presented in subsection 3.4.1. The last design problem considered in the present dissertation is the Design of Experiments (DOE). The DOE entails the investigation of the design space by means of tests – named experiments – executed to assess the response of a design process according to certain design variables believed to influence it.

3.4.1 Fuzzy Logic multi-objective optimization

The main idea at the base of this method is the optimization of the designed airplane through the negotiation and relax of some high level requirements, as range and payload. The negotiation of the product requirements shall be done in accordance with the system's users. Some TLARs might be perceived excessive by the customer. The relaxing of these requirements can bring to an improved (optimal) product, preferable from a user perspective.

This optimization problem can be multi-objective, as the goals of the development process might be, for example in the case of a hybrid powered aircraft, the minimization of the fuel consumption together with the maximization of the safety level (i.e. minimization of the minimum safety altitude). In the context of the doctoral research activities, an optimization method based on the Fuzzy Logic approach is developed [148]. The Fuzzy Logic represents a logic employed to deal with statements that are partially true, i.e. neither completely true or completely false [149]. The Fuzzy Logic has been formalized for the first time by Zadeh in 1965 [150]. Since then, many applications of this new concept of logic have been developed in several fields, mostly regarding control systems. Fuzzy Logic controllers are mainly studied and employed in electronics ([151], [152], [153], [154], [155]). However, control systems based on Fuzzy Logic are spread also in other fields, for example in robotics ([156], [157]), telecommunications [158], air-conditioning applications [159] and medicine [160]. The Fuzzy Logic is rooted in the Aeronautical Engineering, too. Several application studies deal with flight control systems ([161], [162], [163], [164], [165], [166]) but also fire detection systems [167], propulsion systems ([168], [169]) and braking systems [170]. However, other than control system applications, the Fuzzy Logic can be adopted in other contexts. In this regard, a Fuzzy Logic approach has been proposed for multi-objective optimizations in the magnetics field [171]. However, the application of Fuzzy Logic for optimization in aeronautics is still lacking. Moreover, the present subsection proposes an integration between the aircraft multi-objective optimization and the negotiation of the high level requirements.

From a literature review, several multi-objective optimization methods can be disclosed. However, the Fuzzy Logic is considered the most suitable methodology for this kind of problem. The Pareto methods are one of the most well-known multi-objective optimization methodologies. Nevertheless, the Pareto methods allow the determination of a set of optimal solutions all equivalent from a mathematical perspective, without identifying the best one [172]. Furthermore, other two motivations are behind the selection of the Fuzzy Logic approach. Firstly, objectives and requirements defined by quantities measured on different scales can be assessed by means of the Fuzzy Logic. The Fuzzy Logic brings to a transformation of these quantities in scores thanks to which they could be handled together. Secondly, all the design objectives might be ordered following a hierarchy of importance defined by the stakeholders. In other words, the user could retain an objective function more important than another one. Hence, the design process should be led by this more important objective function.

The concept of fuzzy set is at the base of the Fuzzy Logic theory. According to its definition given by Zadeh [150], let *y* being a vector of generic elements of a space of points *Y*. The vector *y* can include requirements of the design case and design results. A fuzzy set *A* in *Y* is characterized by a Membership Function (MF) $\mu(y)$, representing the "grade of membership" of *y* in *A*. The MF $\mu(y)$ associates each point in *Y* with a real number in the interval [0, 1]. In other words, $\mu(y)$ expresses the "degree of satisfaction" of a considered objective as perceived by the customer, on the basis of the value *y* resulting from the design. In the current dissertation, only piecewise linear functions are considered.

By way of example and with reference to Figure 25, let y being one of the objective functions of a certain MDO problem. The customer is supposed to consider unacceptable the values of y resulting from the design if equal or greater to y_{max}. When this parameter results at least y_{max}, a MF $\mu(y_{max})=0$ is set, meaning that the resulting value should be lowered. Vice-versa, results of y equal or below to y_{min} are fully accepted. In this case a MF $\mu(y_{min})=1$ is defined, representing the maximum degree of satisfaction of the customer. Hence, $\mu(y_{min})=1$ and $\mu(y_{max})=0$ state the boundary values of the MF. In the current example, if a solution of the design problem brings to a value of y equal to y*, a degree of satisfaction $\mu(y^*)=0.6$ is returned, as represented in Figure 25. The MF of Figure 25 depicts a decreasing linear function: the degree of satisfaction μ reduces for higher values of y. On the contrary, μ might assume higher results in case of higher values of the objective function, as represented in Figure 26 (a). Additionally, the degree of satisfaction can be characterized by a maximum when y assumes a certain value, as shown in Figure 26 (b).

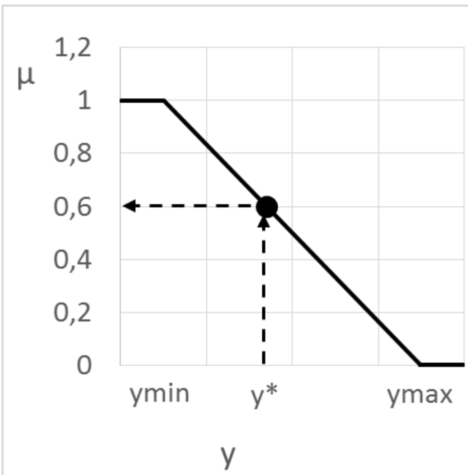


Figure 25: Example of Membership Function of y.

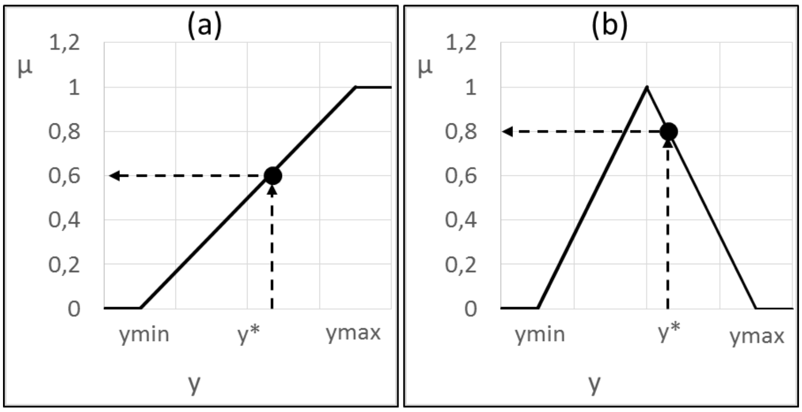


Figure 26: Additional examples of Membership Functions of y.

Once the Fuzzy Logic theory has been applied to all the requirements and objectives of the MDO problem, a global degree of satisfaction is estimated. The global degree of satisfaction combines all the MFs through the intersection of the fuzzy sets ([171], [173]). Given the vector of design parameters *x*, a number *n* of objective functions *y(x)* and relative MFs $\mu_i(y_i(x))$, the global degree of satisfaction *G(x)* results from:

$$G(x) = \min_{i=1,\ldots,n} \{\mu_i(y_i(x))\} \qquad \text{eq. 21}$$

If multiple design problems are performed, each one with different vectors *x* of design parameters, several values of *G(X)* are derived. The optimal solution is the one characterized by the highest degree of satisfaction. This optimal solution can be discovered by an optimizer investigating the design space, looking for the set of design variables constituting the vector *x*, by which the maximum value of *G(x)* is derived.

3.5 Step 4: Implement and verify the design framework

Aim of the step 4 of the development process is the implementation in an executable framework of the MDO problem defined in the previous steps. The integrator is in charge of transposing the MDO problem into the executable framework. Therefore, in the present Section two kinds of design frameworks are presented. The former is a 1st Generation MDO framework (see Section 2.4 for more details about the three generations of design systems), a monolithic framework developed within the context of the doctorate for the conceptual design of an aircraft starting from a set of high level requirements. Even if several design disciplines are encompassed within the proposed MDO system, particular attention is posed on the on-board systems design discipline. This kind of framework can be in fact employed to perform an initial study on the effects of subsystems on the OAD and vice-versa. This preliminary assessment might then bring to the definition of a subsystems architecture, which will be furtherly investigated more in detail with the second kind of MDO framework. The second kind of MDO framework belongs to the third generation. It concerns a distributed multidisciplinary workflow set up within the context of the AGILE project. As this kind of workflow is distributed, the collaborative expert shall intervene in the present step to enable the collaboration among all the disciplinary experts. In particular, the collaborative expert shall develop and deploy tools aimed at supporting the communication among the disciplinary experts and the exchange of inputs and outputs among disciplinary models. For this purpose, the second MDO framework presented in this Chapter will describe tools developed to foster the collaborative design. Again, the current dissertation in focused on the on-board systems design discipline and its integration within the development workflow and interrelation with the other design disciplines.

3.5.1 Example of a first Generation MDO framework

In the current subsection, an example of a 1st Generation MDO workflow is presented. As described in Section 2.4, this type of MDO framework is monolithic, as it is implemented within a single local machine and a unique design team is involved. However, the here proposed 1st Generation MDO workflow is multidisciplinary. Many design disciplines are indeed encompassed during the execution of this example of MDO framework, though the obtained results might be preliminary. However, this workflow is centered on the subsystems design discipline, with the aim of highlighting the impacts of this design discipline on all the other disciplines and vice-versa. In particular, the effects of on-board systems on the entire aircraft level, especially in terms of masses and fuel consumption, are preliminarily investigated.

An example of first generation MDO framework is shown in Figure 27. The workflow presents a convergence MDA problem. The design methodology ([174], [175]) implemented in the present workflow is based on the most renowned L0-L1 fidelity levels design methods found in literature ([68], [104], [70]).

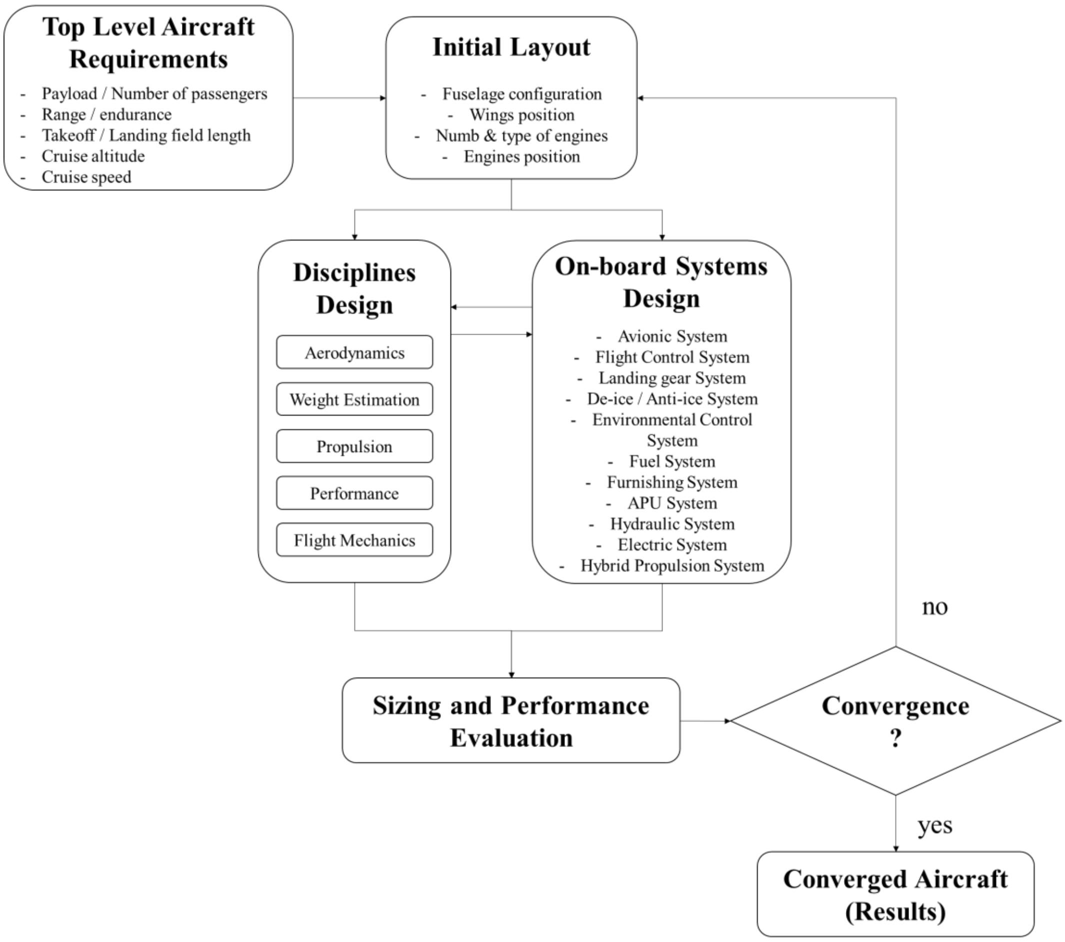


Figure 27: Example of a first Generation MDO framework (adapted from [175]).

The design process starts with the elicitation of the TLARs required by the stakeholders, for instance number of passengers and additional payload, range or endurance, runway length for take-off and landing and a typical mission profile, especially in terms of speeds and altitudes.

Given this set of high level requirements, an initial aircraft layout is sketched. In particular, the fuselage dimensions are determined on the basis of the payload location. Then, the mutual position wing-fuselage is delineated, distinguishing among a low/medium/high wing airplane. At this stage, unconventional layouts might be defined, as either BWB or joined-wing aircraft. Attention is also given on the propulsion system, focusing mainly on type, number and position of the engines. Concerning the particular case of the development of a hybrid-electric aircraft, the hybridization degree is a result of the definition of the initial layout.

Soon after the initial layout definition, the multidisciplinary design of aircraft begins. This multidisciplinary design process encompasses the following design disciplines: aerodynamics, weight estimation, propulsion, performance, flight mechanics and on-board systems. A description of the disciplinary modules is presented in the following subsections, while the on-board systems design module is described in subsection 3.3.2.

Most of the disciplinary methods are based on the MTOW parameter, of which an attempt value is initially hypothesized. The design methodology aims at estimating, at first, the aerodynamic characteristics of the aircraft, as the aerodynamic drag during the mission profile. This trend of the aerodynamic drag is needed, for instance, for the evaluation of the performance during cruise, as the propulsive power or thrust. Furthermore, a first evaluation of the wing area is assessed. This estimation is based on landing requirements, for instance landing distance and Maximum Landing Weight (MLW). From these results, the fuel quantity needed during each phase of the mission profile is calculated, taking into account the typology of propulsion system and relative SFC. The SFC is also affected by subsystems bleed air and shaft power off-takes, as discussed in the description of the "Propulsion" module. The maximum quantity of boarded fuel is finally calculated adding the fuel reserves, generally expressed as a percentage of the total fuel required during the mission. Afterwards, the design process proceeds with a preliminary mass estimation. The masses of the aircraft structural elements (e.g. wings, tail and fuselage), engines and on-board systems are determined. In particular, the masses of the structural components are calculated by means of Weight Estimation Relationships (WERs) developed by the research group at Politecnico di Torino [176]. The propulsion system weight instead is estimated through statistical evaluations based on a collection of several engines present on the market. The on-board systems mass is assessed as described in subsection 3.3.2. Hence, the aircraft empty weight is calculated from the sum of these weights. Taking into account also the fuel weight and the payload, a new value of MTOW different from the tentative one guessed at the beginning of the design process is returned. This result will be the new tentative value of MTOW for the second iteration. This iterative loop will proceed until the design will reach a convergence, i.e. until the difference among the values of MTOW of two sequential loops will be lower than a predetermined threshold.

The first design iteration ends with the selection of the design point. This point represents the optimal one of the infinite design solutions characterized by the particular values of wing loading and thrust loading (or power loading in the case of a propeller aircraft), complying with all performance requirements. More details are provided in the “Sizing and Performance Optimization” subsection.

Aerodynamics

This module entails a very preliminary evaluation of the aircraft aerodynamic characteristics, as the airplane drag during the mission profile. The aerodynamic drag is function of the zero-lift drag coefficient C_{D0}, which can be estimated by means of a statistical formulation provided by Roskam [177]. This statistical formulation considers both the aircraft wet surface and the wing area *S*. The proposed design methodology estimates the value of *S* on the basis of the landing requirements (i.e. airfield altitude, landing distance):

$$S = \frac{2 \cdot W_{LND}}{\rho_{airfield} \cdot {V_{app}}^2 \cdot Cl_{max_{lnd}}} \qquad \text{eq. 22}$$

The MLW W_{LND} [N] can be defined as a percentage of the MTOW [N]. The air density $\rho_{airfield}$ [kg/m^3] depends on the airfield altitude, while the maximum lift coefficient during landing $Cl_{max_{lnd}}$ might be estimated statistically [177]. A first assessment of the approach speed can be done on the basis of the landing distance l_{LND} [m] [177] (see eq. 23).

Given the C_{D0} coefficient, other aerodynamic results might be evaluated, as the maximum ratio C_L/C_D, which is calculated through eq. 24, where the wing Aspect Ratio *AR* and the Oswald factor *f* are inputs of the aerodynamic module that might be obtained by means of comparisons with similar aircraft.

$$\begin{cases} V_{app}[kn] = \sqrt{\dfrac{l_{LND}\ [ft]}{0.5136}} & \text{for CS} - 23 \\ V_{app}[kn] = \sqrt{\dfrac{l_{LND}\ [ft]}{0.507}} & \text{for CS} - 25 \end{cases} \qquad \text{eq. 23}$$

$$\left(\frac{C_L}{C_D}\right)_{max} = \frac{1}{2} \cdot \sqrt{\frac{\pi \cdot AR \cdot e}{C_{D0}}} \qquad \text{eq. 24}$$

Weight estimation

The MTOW is calculated summing three weights: the payload, the fuel weight and the Operating Empty Weight (OEW). The payload is a given high level requirement, while the fuel and the empty weights are evaluated in the present module.

In case of the design of jet aircraft, the fuel weight W_{fuel} [N] is evaluated by means of the following equation:

$$W_{fuel} = SFC \int_0^R \frac{T(x)}{v(x)} dx \qquad \text{eq. 25}$$

The integral of eq. 25 is calculated per each step dx of the total range R [m]. The thrust $T(x)$ [N] might vary along the mission profile as it depends on the aircraft gross weight, which is not constant during the flight (see eq. 36 for the case of the steady level cruise). The flight speed $v(x)$ [m/s] might be constant or it may vary in accordance with the aircraft gross weight. The SFC is expressed in [kg/s/N].

In case of piston engine and turboprop airplanes, eq. 25 might be modified considering a SFC with a different unit of measure [kg/s/kW], and substituting the thrust $T(x)$ inside the integral with the propeller shaft power $P(x)$ [kW].

Concerning the OEW, this parameter is calculated considering structures, engines, on-board systems and miscellanea weight. In particular, miscellanea encompass fight crew, oils, and tools needed for the flight. Concerning the structure weights, several methodologies are available for their estimation, as those reported in [68], [104] and [70]. Methods are also available for the evaluation of the masses of the engines. In the current dissertation, an estimation method based on a statistical base is used. As an example, the graphs in Figure 28 (a) and (b) show a statistical distribution of turboprop and turbofans. From these diagrams, the statistical laws expressed by eq. 26 are derived, obtaining the uninstalled engines mass W_{eng} [kg].

$$\begin{cases} W_{eng} = 0.62 \cdot Max\ Power\ [hp] + 22.7 & \text{for turboprop engines} \\ W_{eng} = 16.5 \cdot Thrust\ [kN] + 301.4 & \text{for turbofan engines} \end{cases} \qquad \text{eq. 26}$$

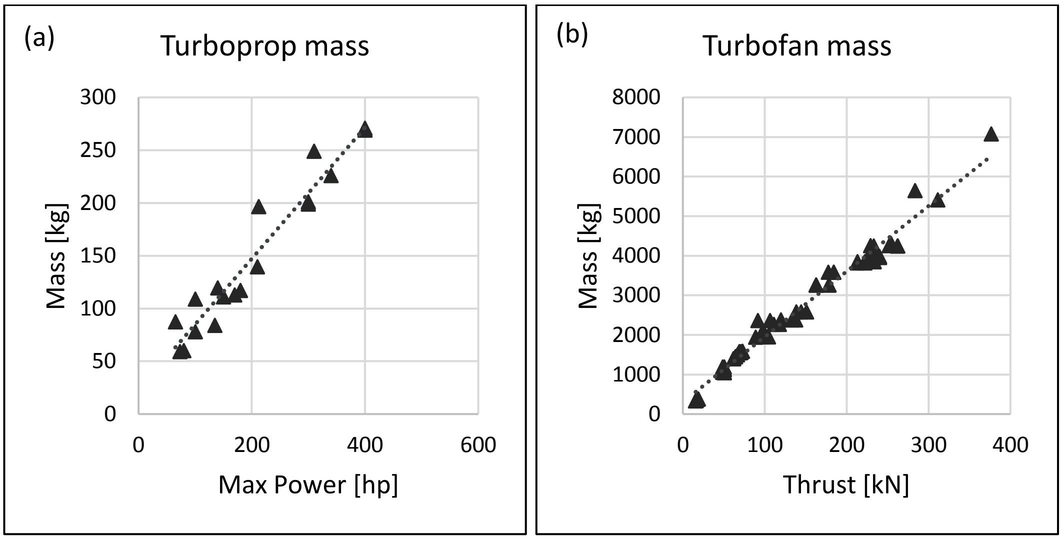


Figure 28: Statistical evaluation of turboprop (a) and turbofan (b) engines mass.

Finally, the mass of the on-board systems is evaluated by means of the dedicated design module – ASTRID – described in subsection 3.3.2.

Propulsion

The main aim of the "Propulsion" module is the evaluation of the engine SFC affected by the on-board systems shaft power and bleed air off-takes. The estimation of the power or thrust required during the mission phases is done instead in the "Performance and Flight mechanics" module.

The present design module receives as input the engine clean SFC (SFC_{clean}), i.e. the SFC considering null the secondary power required by subsystems. The clean SFC value is modified according to the level of on-board systems off-takes. This module assumes relevance from a subsystems design perspective as it entails the comparison in terms of fuel consumption among conventional and innovative on-board system architectures.

The algorithms implemented within the "Propulsion" module bring to a very preliminary estimation of the actual engine SFC. More accurate evaluations should be made by means of higher fidelity engine decks, developed by propulsion experts. A more accurate engine is implemented within the 3rd Generation MDO framework described in subsection 3.5.2.

The methodology coded within the present design module distinguishes among propeller and jet engines. In the first case, eq. 27 might be employed:

$$SFC = SFC_{clean} \cdot \left(1 + \frac{P_{2ary}}{P_{propulsive}}\right) \qquad \text{eq. 27}$$

The power required by the aircraft subsystems is expressed by P_{2ary}, while $P_{propulsive}$ is the power needed for the propulsion. These two values must be expressed with the same unit of measure (e.g. [kW] or [hp]). Analogously, the unit of measure of SFC depends on the clean fuel consumption.

Regarding the jet engines, a methodology proposed by Giannakakis *et al.* ([178]) is implemented within the current module. This method firstly estimates the efficiency of the engine affected by shaft power off-takes (eq. 28) and bleed air off-takes (eq. 29).

The ratio between the engine core efficiencies before (η_{co}^{*}) and after η_{co} the shaft power extraction (P_{po} [kW]) is evaluated by:

$$\left(\frac{\eta_{co}^{*}}{\eta_{co}}\right)_{shaft} = 1 - \frac{P_{po} \cdot \eta_{tr} \cdot \eta_{pr}}{T \cdot V_0} \qquad \text{eq. 28}$$

where η_{tr} and η_{pr} are the transmission and propulsive efficiencies before the extractions of the power off-takes, T [kN] is the engine net thrust and V_0 [m/s] is the aircraft flight speed.

Regarding the bleed air off-takes, the ratio between the two core efficiencies is given by:

$$\left(\frac{\eta_{co}^{*}}{\eta_{co}}\right)_{bleed} = 1 - \frac{2 \cdot W_b \Delta h_b \cdot (BPR + 1)}{(1 - \beta) \cdot T \cdot \left[BPR/\left(\eta_f \eta_{lpt}\right) + 1\right] \cdot (2V_0 + ST)} \qquad \text{eq. 29}$$

The extracted bleed air is expressed by W_b (kg/s), while Δh_b [kJ/kg] is the bleed air enthalpy increase through the core and BPR is the engine By-Pass Ratio. The isentropic efficiencies η_f and η_{lpt} are relative respectively to the fan and to the low pressure turbine and ST (m/s) is the engine specific thrust. β instead is the ratio of bleed air mass flow upon core mass flow, and it is calculated by means of eq. 30:

$$\beta = \frac{1000 \cdot W_b \cdot ST \cdot (BPR + 1)}{T} \qquad \text{eq. 30}$$

Given the efficiency losses due to the shaft and bleed air power off-takes, the ratio of total engine efficiencies assessed before and after the power extractions are calculated by:

$$\frac{\eta_0^*}{\eta_0} = -\frac{V_0}{ST} + \frac{V_0}{ST}\sqrt{1 + \frac{2ST}{V_0}\left[\left(\frac{\eta_{co}^*}{\eta_{co}}\right)_{shaft} \cdot \left(\frac{\eta_{co}^*}{\eta_{co}}\right)_{bleed}\right]\left(1 + \frac{ST}{2V_0}\right)} \qquad \text{eq. 31}$$

This ratio is finally employed for the estimation of the SFC of the engine affected by subsystem power extractions:

$$SFC = SFC_{clean} \cdot \left(\frac{\eta_0}{\eta_0^*}\right) \qquad \text{eq. 32}$$

Performance and Flight mechanics

The thrust or power required during the mission is estimated in the present design module. This module evaluates the thrust or power necessary in take-off, One Engine Inoperative (OEI) condition, climb and cruise.

The engine thrust during take-off T_{TO} [N] is calculated by means of eq. 33 (adapted from [177]):

$$T_{TO} = \frac{22.86 \cdot \left(\frac{MTOM^2}{S}\right)}{l_{TO} \cdot C_{l_{max_{TO}}} \cdot \frac{\rho_{field}}{\rho_{s.l.}}} \qquad \text{eq. 33}$$

The thrust of eq. 33 is function of the wing area S [m^2], the take-off distance l_{TO} [m], the maximum lift coefficient at take-off $C_{l_{max_{TO}}}$, the air density at the airfield altitude ρ_{field} [kg/m^3] and at sea level $\rho_{s.l.}$[kg/m^3].

This module also evaluates the thrust required in take-off in case of one engine failure, in accordance with the regulations CS-23 and CS-25. Therefore, given the number of engines n_e, the airplane MTOW W_{TO}, the $(C_d/C_l)_{TO}$ ratio in take-off and the minimum steady climb angle γ_{OEI} prescribed by the regulations, the needed thrust T_{OEI} in case of OEI condition is calculated through eq. 34:

$$T_{OEI} = \frac{n_e}{n_e - n_e}\left[\left(\frac{C_d}{C_l}\right)_{TO} + \sin\gamma_{OEI}\right] \cdot W_{TO} \qquad \text{eq. 34}$$

Concerning the climb phase, the thrust in climb T_{climb} [N] is evaluated by means of eq. 35, which refers to the schema of Figure 29.

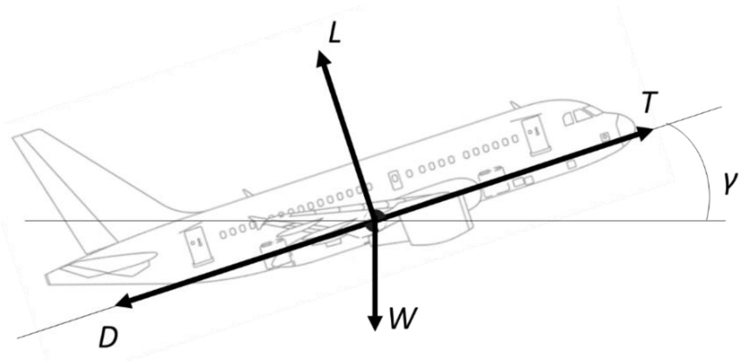


Figure 29: Schema of the forces acting on the aircraft during climb.

$$T_{climb} = D + W \cdot \sin\gamma \approx \frac{1}{2}\rho V^2 S\left(C_{D0} + kC_l^{\,2}\right) + W \cdot \sin\gamma \qquad \text{eq. 35}$$

where D [N] is the drag force in climb, W [N] represents the aircraft gross weight and γ [rad] is the climb angle.

Considering the particular case of steady level flight ($\gamma = 0$), eq. 35 is transformed in order to evaluate the thrust required in cruise T_{cruise} [N]:

$$T_{cruise} = D \approx \frac{1}{2}\rho V^2 S\left(C_{D0} + kC_l^{\,2}\right) \qquad \text{eq. 36}$$

The formulations so far presented may be adapted for the design of piston engine and turboprop aircraft. Considering a propeller efficiency η_{prop} and a flight speed v [m/s], the thrust results of eq. 33, eq. 34, eq. 35 and eq. 36 – here all denoted as T [N] – might be transformed in propeller shaft power requirements, according to the following equation:

$$P_{shaft} = \frac{T \cdot v}{\eta_{prop}} \qquad \text{eq. 37}$$

Sizing and Performance Evaluation

The last module of the 1st Generation MDO framework entails the determination of the design point, which represents a design solution characterized by the optimal values of Wing Loading $(W/S)^*$ and thrust-to-weight ratio $(T/W)^*$ (or power-to-weight ratio). These two values are selected on the basis of the design requirements and results. The minimum value of Wing Loading is evaluated by the ratio of the MTOM and the wing area estimated on the basis of the landing requirements and parameters, as landing distance, MLM, aerodynamic characteristics (i.e. maximum lift coefficient during landing). The optimal value of $(T/W)^*$ is given instead calculating the thrust-to-weight ratio in several conditions, as take-off, one-engine inoperative second segment, climb and cruise.

An example of "Thrust-to-weight vs. Wing loading" chart is depicted in Figure 30.

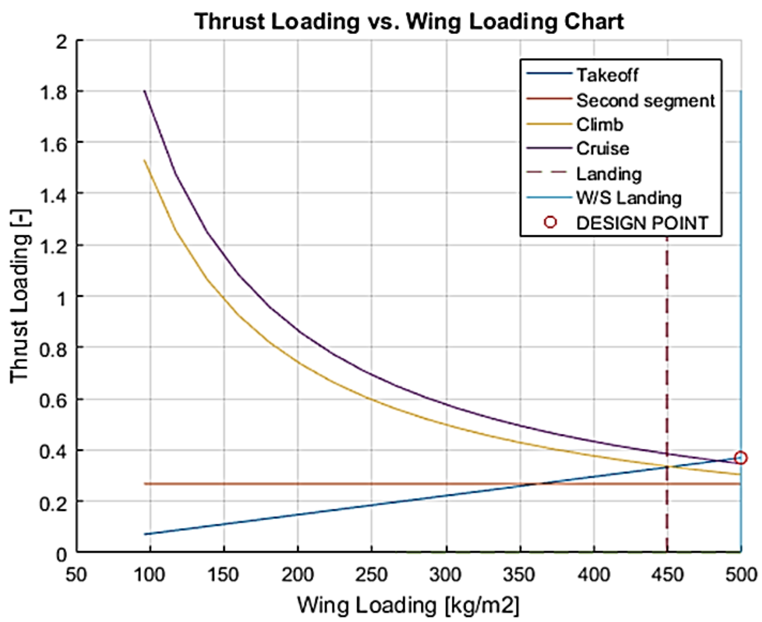


Figure 30: Example of a typical "Thrust Loading vs. Wing Loading" chart.

The graph shows the trend $(T/W) = f(W/S)$ for some mission phases. The vertical light blue line represents the maximum aircraft Wing Loading.

Furthermore, it is worth noting the presence of a second vertical line, representing the ratio *MLM/S*. The design point is represented by the red circle, characterizing the two optimal values of $(W/S)^*$ and $(T/W)^*$.

Example of a Design and Optimization framework

The framework described at the beginning of subsection 3.5.2 represents a design workflow. It is hereafter presented a new framework not only able to design a new airplane, but also to optimize it. The schema of this framework is depicted in Figure 31.

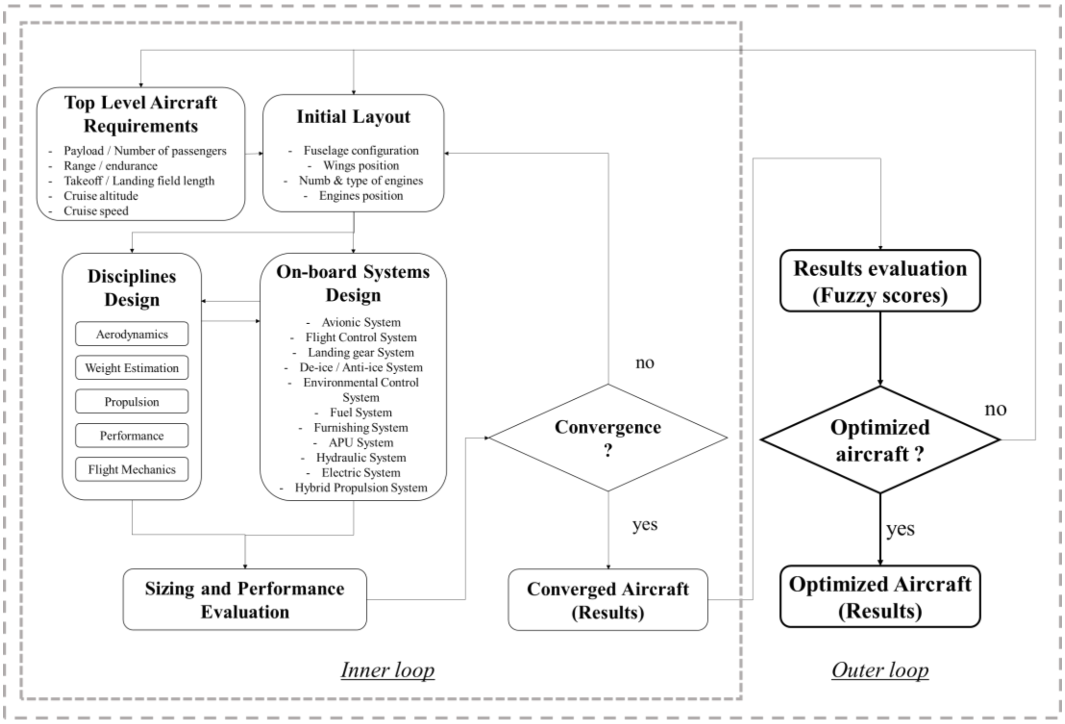


Figure 31: Example of a Design and Optimization framework [148].

The proposed Design and Optimization framework is composed by two parts: an internal part – named "Inner Loop" – and an external one, called "External Loop". The internal loop is conceived to design the entire aircraft, as already described. At the end of this iterative process, a design solution is returned, but this result is not optimized. The definition of the optimal solution is the aim of the "External Loop". In this case, an optimization process based on the Fuzzy Logic is implemented. Some high level requirements and part of the design results are scored by means of Fuzzy Scores, determining the maximum aircraft satisfaction degree, as presented in subsection 3.4.1. Then, an optimizer varies the negotiable high level requirements and some design variables to define the optimal solution, i.e. that one characterized by the highest global satisfaction degree. In particular, a deterministic optimization method based on the Rosenbrock algorithm [179] is selected in the current dissertation.

3.5.2 Example of a third Generation MDO framework

The 3rd Generation MDO framework described in this subsection as an example has been developed within the context of the H2020 AGILE project [180]. This European project encompasses 19 partners from universities, research centers, and industries. Each partner is specialized in a design discipline, as structural design, propulsion system analysis, costs evaluation, optimization processes, collaboration techniques, process integration. The on-board system design discipline – or rather, the subsystem design tool ASTRID – has been integrated within the hereunder described MDO framework in the context of the doctorate. The main objective of the AGILE project is to reduce the aircraft development process – and therefore the TTM – implementing a more competitive supply chain at the early stage of the design [111]. In other words, the ambition of this project is to develop an innovative multidisciplinary design framework, integrating advanced design and optimization techniques and enhancing the collaboration of several experts with different skills, backgrounds and affiliations. One of the main innovations of the AGILE 3rd Generation MDO framework is represented by the inclusion of the subsystems design. Therefore, the hereunder described framework can be also employed for the assessment of subsystem impacts on the other design disciplines since the conceptual design phase. In particular, applications of the AGILE MDO framework proposed in Chapter 4 will show some of these impacts on the OAD, mainly due to masses and power offtakes of different on-board system architectures.

The main elements of this kind of innovative MDO environment are three. The first one is an engineering framework software for the management of the development process and the optimization. This type of tool is named Process Integration and Design Optimization (PIDO) environment. In particular, two kinds of PIDO software are employed in AGILE. The first one is the “Remote Component Environment” (RCE) [181] developed by the German Aerospace Center DLR. The second commercial tool is “Optimus” framework [182], provided by NOESIS Solutions.

The second main element required for a 3rd Generation MDO framework is a common namespace for the exchange of information between the disciplinary experts, hence supporting the collaboration among different experts. This central data model is represented by CPACS (see subsection 1.3.1).

The last element is represented by the disciplinary tools. This modules should be able to extract the required information from the CPACS, and then upload the obtained results. Furthermore, the disciplinary modules shall be implemented within a PIDO framework, in order to connect them together in a single design process. An example of disciplinary tool concerning the preliminary design of aircraft subsystems is presented in subsection 3.3.2.

Actually, an additional element is required to enable the interconnections among the disciplinary tools. For this purpose, a software named Brics has been coded by the Dutch Aerospace Center NLR [183]. Brics encompasses all the technologies required to support the realization of cross-organization collaborative workflows, for instance complying with companies IT security constraints.

More details concerning the MDO framework developed in the AGILE project and about the integration of the aircraft subsystems design module within this collaborative framework are presented in the following subsection.

The AGILE framework

In Figure 32 is depicted an example of a 3rd Generation MDO framework. It represents an OAD process implemented within the AGILE project [119].

From the figure it is possible to note the cross-organizational, distributed and multi-disciplinary aspects of the proposed design framework. A few design disciplines are involved within this design process. These design disciplines are analyzed by different European, Russian and Canadian partners distributed in several locations.

Among all the disciplines encompassed inside the proposed framework, in the current dissertation the attention is focused on the aircraft subsystems preliminary design. In particular, the integration of this discipline within the MDO environment is hereafter described. Case studies and results concerning the framework of Figure 32 are instead proposed and discussed in Chapter 4.

Figure 33 shows the implementation of the subsystems preliminary design tool, ASTRID, integrated within the PIDO software Optimus. This version of ASTRID is coded in Matlab language and it fulfils two prerequisites:

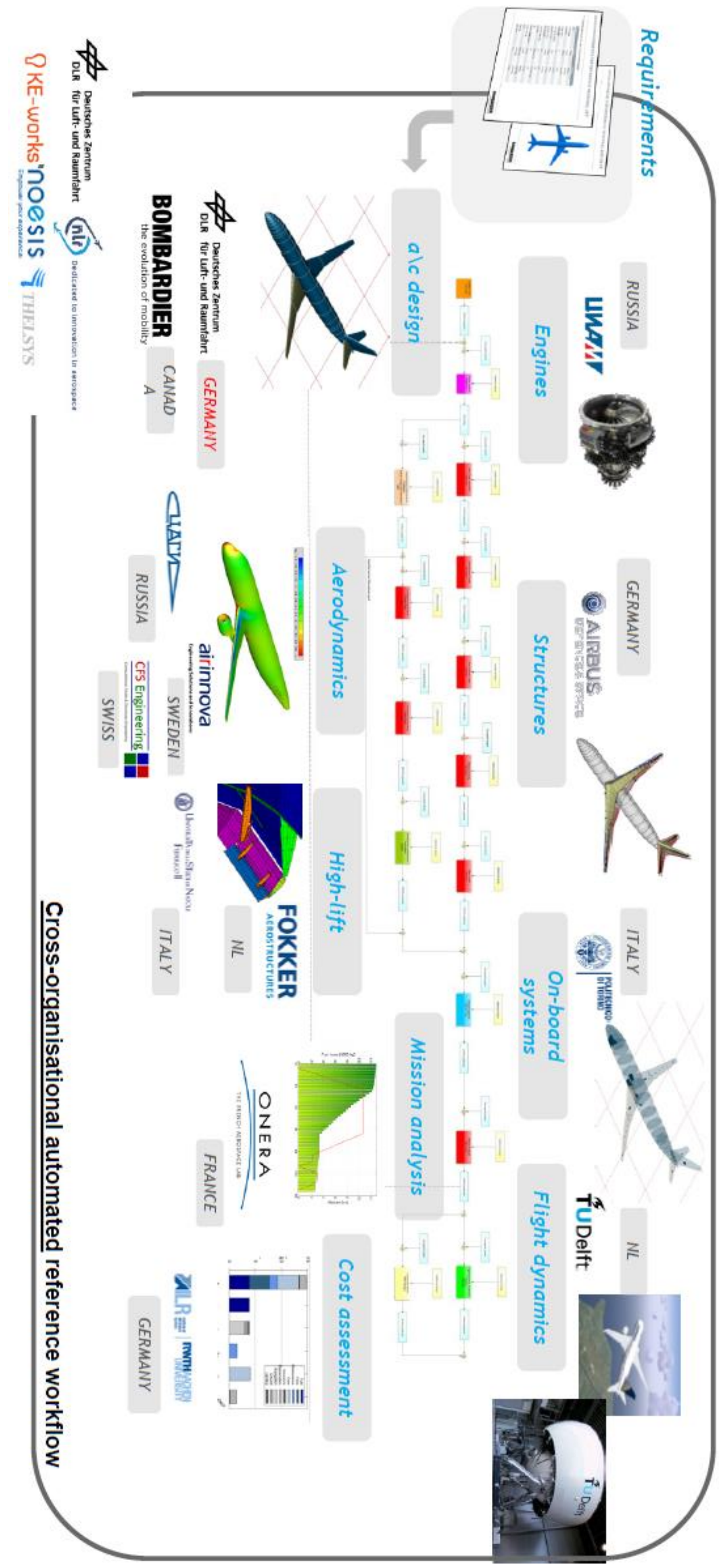


Figure 32: Example of a third Generation MDO framework [119].

1) It runs without any user interaction. An MDO process might be characterized by thousands loops and iterations. Thus, all the required tool inputs shall be provided automatically once evaluated from the other design disciplines, or must be initialized before the tool execution.
2) It is executable from command line. This command is automatically given by the workflow instead of the disciplinary expert.

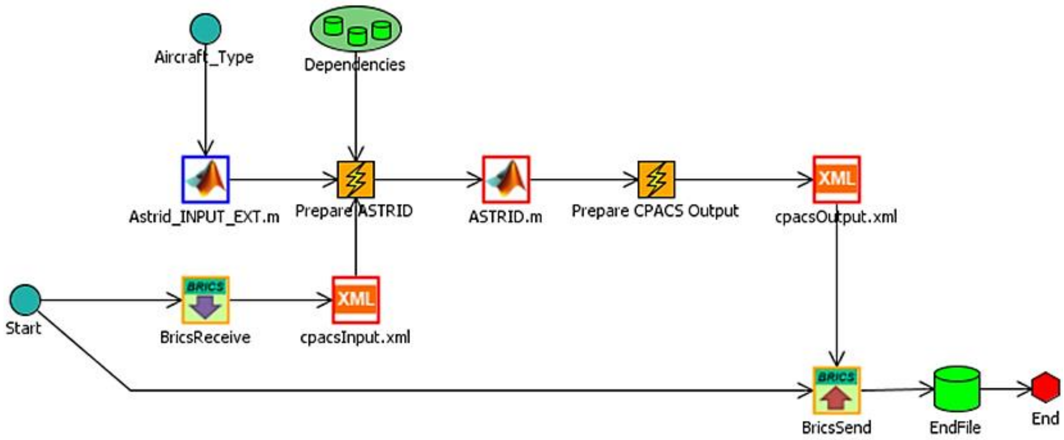


Figure 33: Implementation of ASTRID within the AGILE MDO framework.

The prerequisites compliant version of ASTRID implemented within the framework receives two input files. The first one is represented by the CPACS input xml file. It is updated on the basis of the results obtained during the previous disciplinary analyses. For instance, the primary surfaces hinge moment evaluated by the aerodynamic tools is employed by ASTRID for the sizing of the actuation systems. The second input file, named “Astrid_INPUT_EXT.m”, contains all the specific parameters peculiar of the subsystems design discipline, for instance types of electric voltage and hydraulic system pressure. These values are not provided by any disciplinary module of the workflow, but it is the responsibility of the on-board systems expert to define them.

Once ASTRID has completed the execution and the results from this analysis have been derived, the CPACS file is updated with the new outputs and then passed to other disciplines. Both the input and the output files are exchanged with the other disciplines by means of “Brics” software. The CPACS files are stored inside a server hosted at the DLR of Hamburg, in Germany. From the workflow of Figure 33 it can be seen that two “Brics” interfaces are integrated to download and upload the two CPACS files.

An example of application of a workflow analogous to the one depicted in Figure 32 will be provided in Section 4.4. In particular, other than the subsystems design tool, this version of 3rd Generation MDO framework encompasses the following main disciplinary modules:

1) <u>Aircraft synthesis</u>: the module includes aerodynamic and structural analyses. It is implemented within the tool VAMPzero owned by the German research center DLR ([51], [184]). The main aircraft aerodynamic characteristics (e.g. the drag polar) are computed through a VLM model based on the well-known AVL solver. Furthermore, the Aircraft Synthesis module calculates structural loads and perform FEM based structural analyses.
2) <u>Low speed aerodynamics</u>: both University of Naples and the Swiss company CFS Engineering provide expertise regarding the aircraft aerodynamics, focusing on the detailed design of the high lift devices.
3) <u>Propulsion</u>: the Russian Central Institute of Aviation Motor Development (CIAM) is in charge of modelling the propulsion system. Preliminary results including engine sizing and performance are obtained by means of the commercial tool GasTurb v12 ([87], [88]).
4) <u>Nacelle and airframe integration</u>: the aerodynamic characteristics of the nacelle and its integration with the airframe are investigated by Russian institute TsAGI [185]. Firstly, the aerodynamic analysis of the isolated nacelle is investigated according to the ambient flow, engine geometry and engine gas dynamics properties. Then, the coupling influences among the nacelle and the airframe are investigated.
5) <u>Mission performance</u>: the aircraft performance relative to each mission phase is computed by a module developed by DLR. This tool requires input concerning the airplane aerodynamics, the engine performance, the aircraft weights to estimate the block and reserve fuel required during a predefined mission.
6) <u>Cost analysis</u>: the LCC of the aircraft is assessed by means of a simulation tool developed by the Institute of Aerospace Systems of RWTH Aachen University. Semi-empirical methods described in [186] and [187] are implemented within the module for the evaluation of both recurring and non-recurring costs.

3.6 Step 5: Create and select the design solution

The last step of the development process entails the execution of the MDO framework implemented in the fourth step. One or more solutions are obtained by running the workflow. The design team is responsible for the assessment of the obtained results. The architect shall inspect the overall process, while each disciplinary expert shall evaluate the results of its own discipline. Moreover, the obtained solutions must be compliant with all the high level and derived requirements elicited and developed in the first step of the development process. The customer must be involved for the validation of the design solutions and compliancy with all the required product functionalities.

The solutions obtained at the end of the development process might define an initial baseline. This baseline can be further and deeper designed and optimized in following iterations. Otherwise, lower level problems can be considered, as the development of components. In these cases, the five-step development process might be again employed, perhaps including higher fidelity disciplinary codes.

The development methodology proposed in this Chapter is used for the preliminary design and optimization of four case studies, which are presented and described in Chapter 4. In particular, the algorithms, design techniques, methods and design frameworks previously described are utilized in the case studies.

Before concluding the present Chapter, it is hereunder proposed an integration of the DTC approach within the described MDO process. This approach would cover all the five steps of the methodology, aimed at eventually deriving the optimal solution compliant with a target cost requirement.

3.7 Integration of a Design-To-Cost approach within the MDO process

As explained in Section 2.3, the DTC is a design approach aimed at reducing the aircraft development and production costs. A six-step methodology is proposed in the present dissertation for the integration of a DTC approach within the collaborative MDO process. In particular, the proposed methodology aims at performing a trade-off study among different innovative concepts (i.e. aircraft or system architectures) derived from a conventional solution. Moreover, the proposed methodology aims at integrating the aircraft functional and performance design. In this regard, a “Value Engineering” approach [188] is adopted to score the various solutions and to select the one compliant with the target cost and characterized by the higher "value". The Value Engineering is a collection of methodologies and techniques aimed at designing the product with all the capabilities required by the customer at the lowest development cost. Thus, the Value Engineering consists of identifying and removing all the functionalities of the product that entail augments of the costs even if are not required by the customer. Hence, the parameter "value" is defined as the ratio:

$$Value = \frac{Importance\ of\ capabilities}{System\ cost} \qquad \text{eq. 38}$$

where the importance of the capabilities can be expressed by a subjective score defined by the customer.

The proposed methodology encompasses the following six steps:

1) Cost estimation of the conventional solution: an analogy or parametric model (see subsection 2.3.1) can be adopted for the cost estimation of the conventional solution. This cost derives from the results of the functional design (step 1) and the performance design (steps 3, 4 and 5).
2) Definition of the target cost: the target cost of the innovative product shall be assumed on the basis of its target price, as explained in Section 2.3. Furthermore, the conventional solution cost previously estimated might represent a reference value for the new product target cost.
3) Identification of the alternative innovative solutions: from a functional analysis performed during the first step of the MDO process described in the present Chapter, several alternatives of the product can be identified. These solutions would be characterized by different capabilities and functionalities. Each one of this solution should be scored on the basis of interviews with the involved stakeholders. More specifically, the numerator of eq. 38 should be properly assessed.
4) Preliminary design of the alternative solutions: the five-step MDO process shall be set up and executed for design of all the alternative solutions
5) Cost estimation of the alternative solutions: as done for the conventional solution, the costs of all the alternatives are estimated on the basis of the results obtained from the previous point.
6) Selection of the optimal innovative solution: the optimal solution shall be selected according to its cost and its "value" result. This solution must be indeed compliant with the target cost determined in 2). If the costs of all the

solutions exceed the target cost, other alternatives shall be identified, repeating the proposed approach from 3). Whether more than one solution are compliant with the cost requirement, the selected one should be characterized by the highest “value”, although with a higher cost.

The fourth case study presented in the following Chapter will demonstrate the potentialities relative to the adoption of the DTC approach within a MDO process.

Chapter 4

Design case studies

4.1 Introduction

As claimed in Section 1.4, one of the two main objectives of the present dissertation is the preliminary design of conventional and innovative aircraft on-board systems. In particular, the attention is posed on the development of hybrid-electric propulsion systems and More Electric and All Electric subsystem architectures. The design and optimization of these new solutions is indeed one of the main contribution of the doctoral dissertation.

This Chapter presents four cases studies of the proposed design methodology. The four applications cover all the five steps of the methodology described in Chapter 3. In particular, the first objective of Chapter 4 is the validation of the ensemble of design techniques and sizing algorithms proposed in the previous Sections. The second objective is the demonstration of the effectiveness of the proposed methodology on the basis of the two main motivations of the present doctoral research:

1) Development of better products: higher performance, lower operating costs, lower environmental impact;
2) Enhancement of the competitiveness among aeronautical industries, in particular focusing on the enhancement of the development process: reduction of TTM (and hence reduction of the duration of the design process), reduction of the development and production costs, minimization of the risks due to wrong design choices.

The first application study (see Section 4.2) is focused on the first step of the design methodology. The case study targets the functional development of a hybrid propulsion system. The MBSE approach is adopted to derive all the system capabilities effectively required by the customer and all the involved stakeholders. Performing this task during an early stage of the design process, many architectural and layout choices can be taken in compliancy with the stakeholders needs, minimizing wrong design choices that might entail further modification costs.

The second case study (see Section 4.3) is again centered on the development of hybrid propulsion systems. However, in this second application the attention is not focused only at the subsystem level, but also at aircraft level. In other words, developments of whole hybrid aircraft are done, using as reference a general aviation airplane. The main aim of this case study is the design of more environmentally friendly products, but attention is also focused on their performance characteristics, in particular with the purpose of enhancing the safety level of single-engine general aviation aircraft.

The development of products characterized by lower operating costs is the objective of the third application study (Section 4.4). The peculiarity of this case study is represented by the design of innovative on-board system architectures. According to the literature review ([97], [189], [96], [135]), these innovative architectures should entail lower engine fuel consumptions, with the main aim of reducing the aircraft operating costs. Both 1st and 3rd Generation MDO frameworks are employed in this case study, with the objective of accelerating the development process. In particular, the MDO frameworks described in subsection 3.5.1 and subsection 3.5.2 are employed.

The last application study (Section 4.5) is based on the DTC approach, as its main motivation is the reduction of the development and operating costs. All the five steps of the proposed methodology are treated in this case study, from the elicitation of the high level requirements (Step 1) to the execution of the implemented MDO framework (Step 5). The last application study is again centered on the development of hybrid propulsion systems.

Limitations of the case studies

Although the design case studies presented in the current Chapter represent one of the main contribution of the dissertation since innovative hybrid and More Electric aircraft are treated, some limitations are present and hereunder highlighted. In particular, three main limitations of the case studies are identified.

First, uncertainties are not taken into account. Uncertainties in the design process are generated by assumptions, limitations of the design models, incomplete knowledge on the application case. Furthermore, uncertainties highly affect the design of innovative solutions due to their scarcity of data and information. Bandte [190] discerns three sources of uncertainty impacting on the aircraft design process. The first source of uncertainty regards the operational environment in which the product will be employed. Political, social and economic reasons of the coming years or decades might question design decisions taken at present. Moreover, projections into the future of the current technologies introducing enhancements also affect the design process. Finally, the third type of uncertainty is represented by the analysis models themselves. Models are inevitably only approximations of the real physical situations. Due to the early design phase, the low-medium fidelity and accuracy levels of the employed models indeed affect the outcomes of the analyses. Furthermore, all these sources of uncertainty cause error propagations among all the design disciplines. The uncertainties concerning the results of the subsystem design discipline would impact on aircraft performance, masses, dimensions and fuel consumption. Therefore, sensitivity analyses shall be performed to quantify the magnitude of the impacts of all the types of uncertainty at discipline and aircraft levels. Several studies of this challenging topic are present in literature, also addressed to the conceptual design of on-board systems (e.g. in [190], in [21] and in [66]). In particular, it is worth mentioning the analysis performed by Chakraborty *et al.* [191] about the assessment of the second and third types of uncertainty previously described on different aircraft subsystem architectures. It is therefore clear the need of detailed studies on this topic, especially in the case of initial design of on-board system unconventional solutions. In conclusion of the description of the present limitation, it is worth noting that all the results reported in the present Chapter – especially those concerning innovative subsystem architectures – are contained within confidence intervals. However, the confidence intervals themselves are uncertain. Therefore, the proposed case studies aim at highlighting the possible evolutions of the design solutions according to the introduced innovations (e.g. increment/decrement of aircraft weights).

The second limitation regards the impossibility of validating the design models for the development of innovative solutions. The validation of a model is done comparing the results obtained from the analysis with the real data available in literature. This can be done in case of design of conventional solutions, as several information of different aircraft is publicly available. However, data of innovative solutions is missing, mainly due to two reasons. Firstly, only a few innovative aircraft have been developed so far. The literature review provided in Section 1.2 describes some of the few hybrid and More Electric aircraft currently operative. Secondly, data of current innovative solutions is confidential and hence still secret. Although some pieces of information can be gathered from the literature – e.g. the amount of weight saving due to the innovative architecture of the Airbus A380 (see subsection 1.2.1) – more data is required to make a detailed comparison among the obtained results and the real aircraft specifications. Therefore, the following procedure is adopted in the current dissertation. The development process starts from the initial design of a conventional solution, which can be validated. The conventional solution then represents the starting point of the design of innovative architectures. The innovations of the unconventional solutions are indeed enhancements of the conventional one. Furthermore, great emphasis shall be gave to the comparison among all the alternative solutions, instead of focusing on the results assessed alone.

The last limitation regards the quantification of the on-board systems design effects at aircraft level. In Section 3.3.1, a qualitative overview of the effects of the subsystem design discipline on the other disciplines was provided. However, due to the limitations of the analysis tools implemented within the 1st and 3rd Generation MDO frameworks described in Section 3.5, several of these subsystem effects at aircraft level are neglected in the present dissertation. For instance, the impacts of the bleedless subsystem architecture on the aerodynamic drag and hence on the fuel consumption, aircraft masses and engine thrust are not quantified. The involvement of additional and different competence experts would definitely bring to a deeper quantification of all the on-board systems design impacts on the OAD.

4.2 Conceptual functional design of a Hybrid Propulsion System

As case study of the functional design employing the MBSE approach described in subsection 3.2.1, the conceptual development of a HEPS is presented below.

This propulsion system is conceived to be installed aboard a Medium Altitude Long Endurance (MALE) UAV. This unmanned aircraft should be designed for ISR (Intelligence, Surveillance and Reconnaissance) tasks. This case study has been carried out within the context of an Italian project, named TIVANO (Tecnologie Innovative per Velivoli di Aviazione Generale di Nuova GeneraziOne – Innovative

General Aviation Technologies). This project has been funded by the Italian Ministry of Education, University and Research, and was aimed at assessing and developing innovative technologies to be applied on general aviation aircraft and Remotely Piloted Aerial Systems (RPASs). One of the investigated technologies was the design of hybrid propulsion systems for these classes of aircraft, and the evaluation of their benefits and drawbacks.

The propulsive power shall be generated by two different kinds of source: one endothermic and the latter electrical. This “hybrid mode” is required during the only take-off phase, while in taxi a zero-emissions propulsion is requested. During the other mission phases the propulsion system should be traditional: only the endothermic source shall provide propulsive power. The last high level requirement concerns the emergency: the propeller moved by the airflow might recover energy to supply the on-board users or to generate propulsive power at a later stage, when needed. This feature is called “Ram Air Turbine (RAT) mode”.

The Table 9 collects the high level requirements at aircraft level, from which the requirements of the propulsion system are derived.

Table 9: Air-Vehicle and Propulsion System Level Requirements.

Air-Vehicle Level	**Propulsion System Level**
The air-vehicle shall perform the green taxi	During taxi the system shall generate thrust with only the electrical source
	During taxi the system shall generate electric power by the electrical source to supply the electric users
	During taxi the system shall have electric energy stored for the primary power
	During taxi the system shall have electric energy stored for the secondary power
In case of emergency the air-vehicle shall generate electric power to supply users	In case of failure of the endothermic source the system shall receive energy from the external environment (RAT mode)
	In case of failure of the endothermic source the system shall store electric energy received from the external environment
In case of emergency the air-vehicle shall reduce the rate of descent	In case of failure of the endothermic source the system shall generate propulsive thrust by the electrical power source
	The system shall be equipped with a endothermic source and an electrical source

The air-vehicle shall be equipped with an hybrid-electric propulsion system	The system shall supply electric power to the EPDS
	The system shall generate thrust necessary for the taxi
	The system shall generate thrust necessary for the take-off
	The system shall generate thrust necessary for the flight
	The system shall accept fuel from the fuel system
	The system shall store electric energy (to be used as primary power source)
The air-vehicle shall control each subsystem	The system shall automatically manage the power splitting between the two sources
	The system shall automatically manage the power switching between the two sources
	The system shall automatically manage the level of power of the two sources
	In case of emergency the system shall disconnect the electrical non-essential users
	The system shall maximize the efficiency for the generation of the propulsive thrust in each phase of flight
	In case of emergency the system shall manage the status of the electrical source in order to maximize the range

From the system requirements, four UCs are derived, as represented in the UC Diagram of Figure 34.

Provide propulsive power

This UC persists during all the mission of the aircraft, from the taxi out before the take-off phase to the taxi in after landing. The system should provide propulsive power in three different modes. An “only-electric” mode during the taxi phases and in case of failure of the endothermic source when the airplane is airborne. A “hybrid” mode, consisting of the combination of mechanical power generated by the endothermic source and by the electrical source, allowing a Powerboost during phases – such as the take-off – when a surplus of thrust is required. A “traditional” mode, in which the propulsive power is supplied only by the endothermic source.

Provide secondary power

Similarly to the previously described UC, the “Provide secondary power” UC persists during all the mission from and to the taxi phases before and after the flight. The system should provide the aircraft users with electrical power, even in case of failure of the endothermic source.

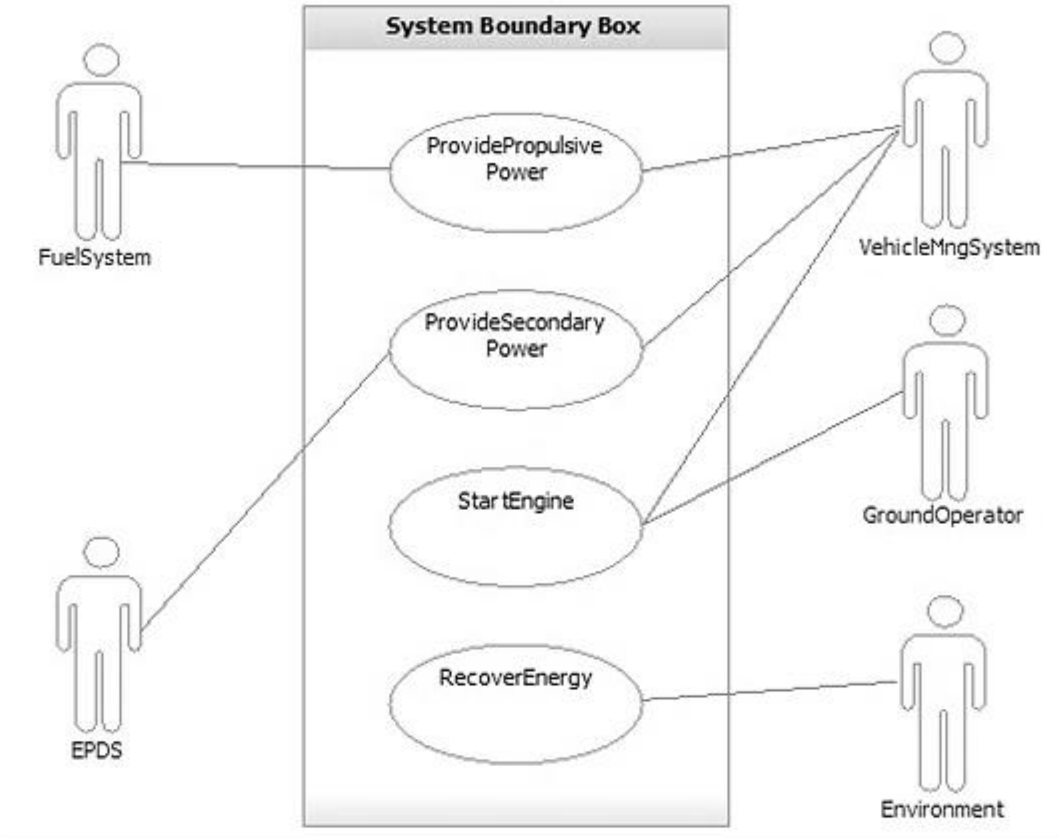


Figure 34: UC Diagram of the Hybrid Propulsion System.

<u>Start Engine</u>

The “Start Engine” UC may occur on ground, as nominal situation, and in-flight, after an engine shut down. In the first case, a ground operator starts the engine. In case of engine failure during flight, the system should perform a predetermined number of engine starting attempts, before establishing the definitive loss of the endothermic source.

<u>Recover Energy</u>

The last UC refers to the energy recovery by means of the propeller during the descent. This energy might be used to supply the electrical users or to produce propulsive power when needed.

Within the context of the functional development, five operating modes of the HEPS are defined:

1) Powerboost: both the endothermic and the electric sources are employed for the generation of propulsive power. This operating mode occurs when the surplus of power is required, such as during take-off. Electrical users shall be supplied by electric energy storages.
2) Traditional: when lower values of power are required, such as in cruise condition, only the endothermic source is active. This source provides both propulsive and secondary power.
3) GreenTaxi: the system should perform a zero-emission taxi, generating propulsive power only from an electric source. During the taxi, the propulsion system should provide the electrical users with electric power by means of energy storages.
4) RATmode: the propeller can be employed as RAT, transforming the external airflow energy into electric energy, hence supplying the electrical users and charging an electric storage system. This can be done during descent to recover energy or in case of a malfunction of the endothermic source, therefore enhancing the safety level of the aircraft.
5) ElectricFlight: the energy stored during the RAT operating mode might be used feeding the propeller, hence smoothing the descendent trajectory when needed. In this case, part of the stored electrical energy is used to supply the electrical users.

Following the IBM Harmony methodology, the development of the hybrid propulsion system proceeds with the System Functional Analysis, a UC based study in which the functional requirements are transformed in a coherent description of the System Functions. This task is performed for each UC considered in the design. However, only the UC “Provide propulsive power” is here treated as example. Therefore, in Figure 35 is shown the Black-Box Activity Diagram of this UC.

The activity starts with the request of the throttle command from the Vehicle Management System (VMS) to the propulsion system. The VMS is an on-board computer for the management of navigation, aircraft subsystems and emergencies. If the operating mode is “Traditional”, the fuel system should provide the propulsion system with fuel; instead, if the operating mode is “ElectricFlight” or “GreenTaxi”, the system should receive and transfer electric power from the propulsive battery to the electric source. In case of “Powerboost” operating mode, the propulsion system should concurrently require fuel and transfer electric energy from the storage system to the electric source. The resulting mechanical power – from the endothermic or the electric source or from both – is finally transferred to the propeller, hence generating propulsive traction.

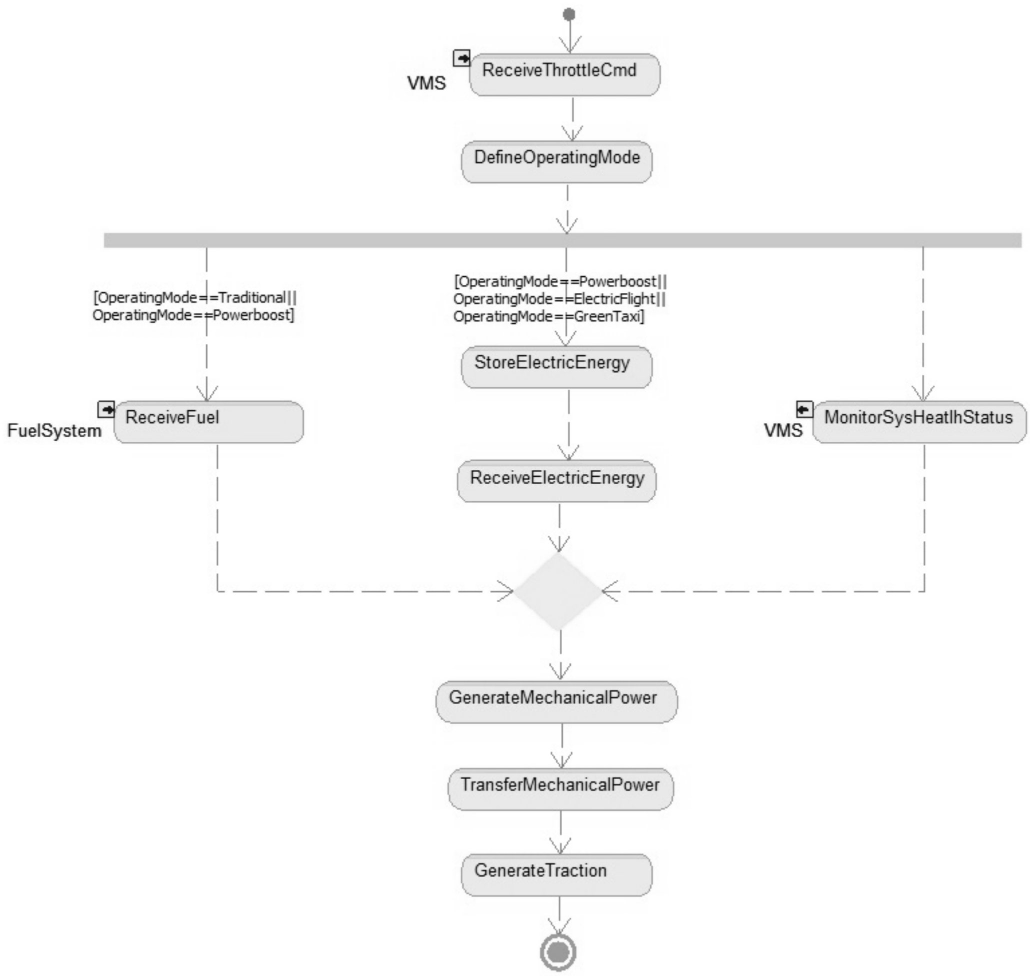


Figure 35: Black-Box Activity Diagram of the UC "Provide propulsive power".

The Black-Box Sequence Diagram of the current UC, showing the list of actions performed by the system and the messages exchanged with the external users – in this case the VMS and the fuel system – is depicted in Figure 36.

The System Functional Analysis ends with the realization of the Statechart Diagram, which represents the states of the systems. The diagram relative to the UC "Provide propulsive power" is depicted in Figure 37. The Statechart Diagram validates the logic behavior of the system according to the stakeholders' requirements.

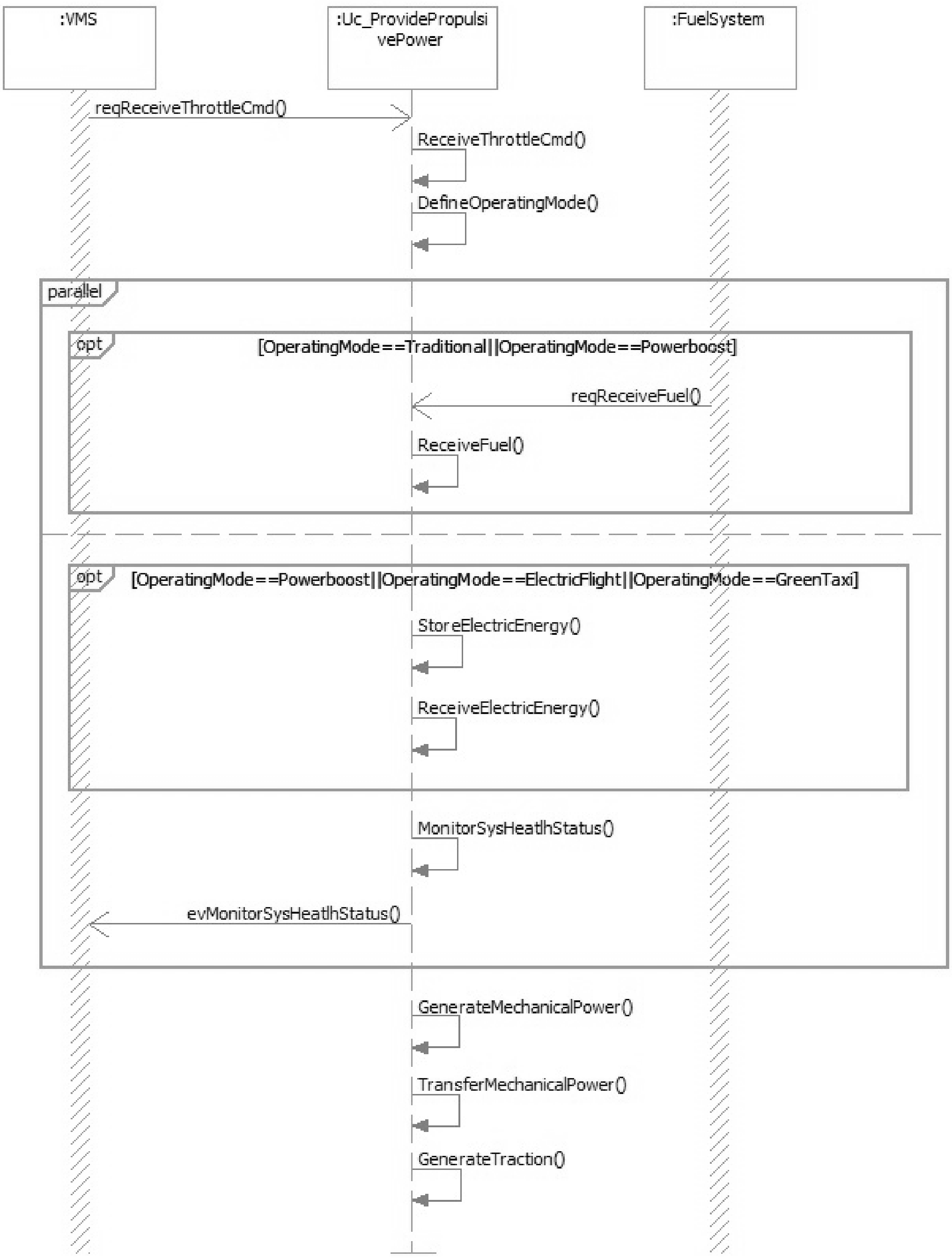


Figure 36: Black-Box Sequence Diagram of the UC “Provide propulsive power”.

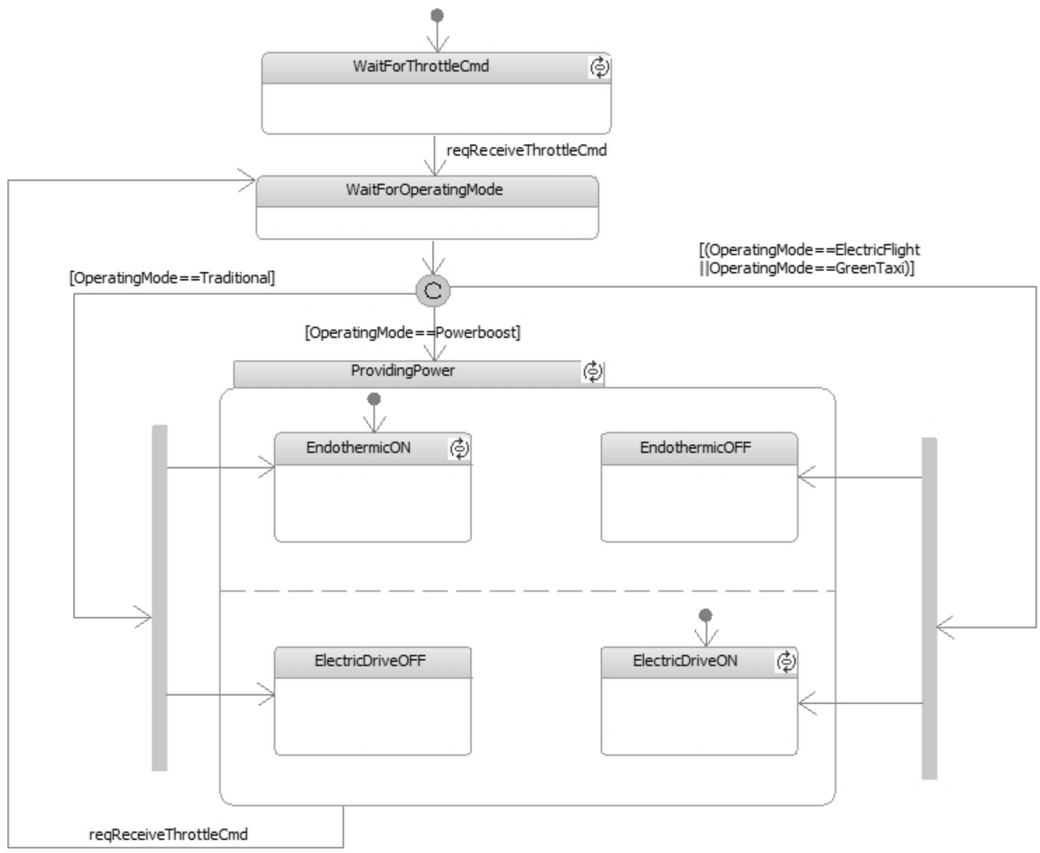


Figure 37: Statechart Diagram of the UC "Provide propulsive power".

At the beginning the system waits for a throttle command and for the operating mode. If the operating mode is set to "Traditional" (Figure 38 (a)), the system passes to both the states "EndothermicOn" and "ElectricDriveOff": the endothermic source is the unique one that provides propulsive power. Otherwise, if the operating mode is equal to "Powerboost" and the energy storage is not discharged (Figure 38 (b)), both the endothermic and the electrical source supply mechanical power to the propeller. In this case the active states are "EndothermicOn" and "ElectricDriveOn". Finally, if the operating mode is "ElectricFlight" or "GreenTaxi" (Figure 38 (c)), the propulsive system passes to the states "EndothermicOff" and "ElectricDriveOn". When a new throttle command is received, the states are updated and modified according to the new operating mode.

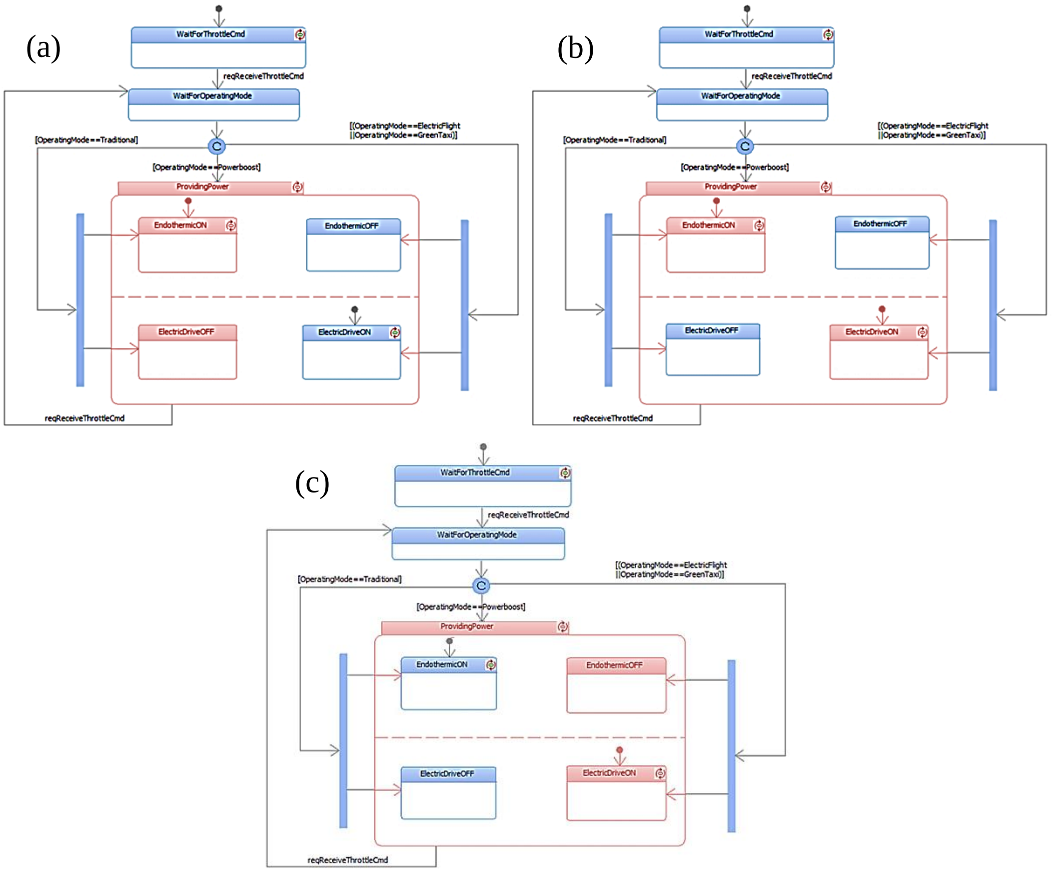


Figure 38: Different states of the system.

In the last phase of the IBM Harmony methodology, the "Design Synthesis", the physical architecture of the hybrid propulsion system is defined, taking into account all the required functions.

Concerning the present case study, the first stage of the "Design Synthesis", the Architectural Analysis, is neglected, passing directly to the Architectural Design. In this sub-phase the system is decomposed in parts, each one representing a component (e.g. ICE, EM, batteries). In Figure 39 is reported the BDD of the propulsion system, in which all the main components of the system are represented.

Therefore, the hybrid propulsion system is composed by the following components:

- A propeller.
- A mechanical transmission connecting the thermal engine, the electric motor/generator and the propeller.

- A joint connecting the ICE with the propeller.
- A piston engine.
- An electric machine employed as both electric motor and electric generator.
- A battery with relative Battery Management System (BMS) required to supply electric power for the propulsion during the hybrid and electric modes.
- A system controller used to manage all the equipment according to the operating modes and the status of the components.
- A cooling system, required to protect the equipment from heating.

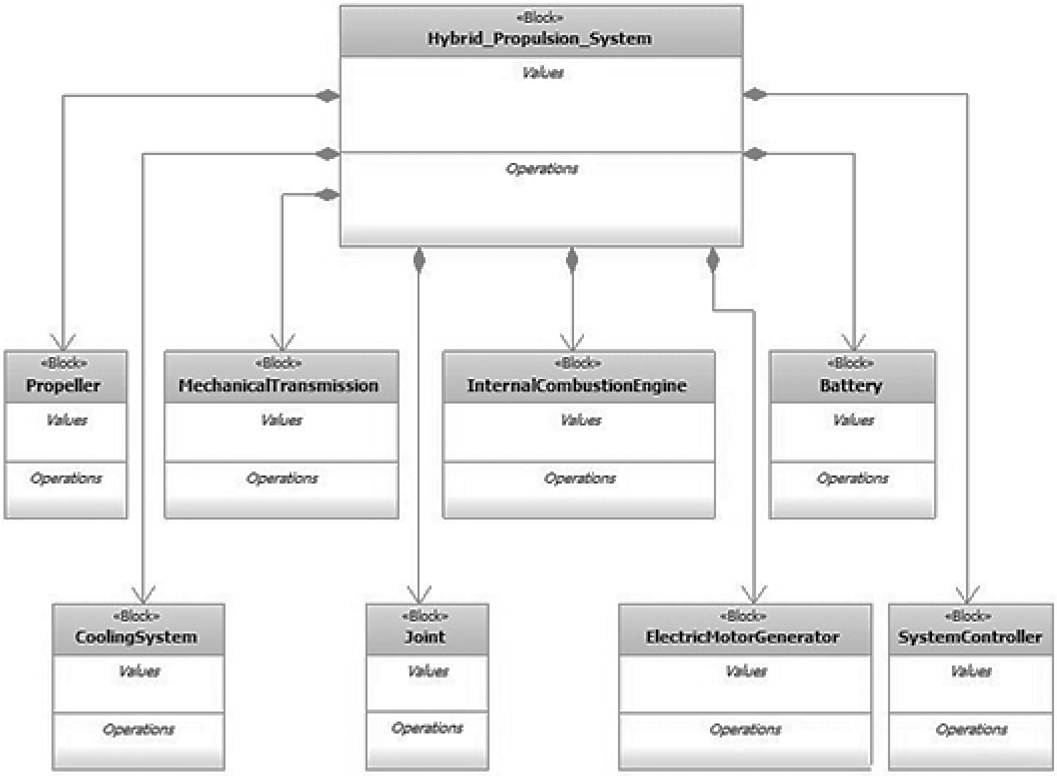


Figure 39: BDD of the Hybrid Propulsion System.

Thus, all the operations are allocated to the parts, as partially shown in Table 10, regarding the only “Provide propulsive power” UC.

As previously stated, one of the advantages of the MBSE approach is the exploitation of the models in further projects and design case. In the last case study presented in the dissertation (Section 4.5), the functional model just described will be employed in the definition of several alternative solutions of hybrid propulsion system, each one characterized by the removal of some capabilities, as the GreenTaxi functionality.

Table 10: Functions allocation to system components.

Component	Allocated functions (“Provide prop. power” UC)
System controller	ReceiveThrottleCmd; DefineOperatingMode
ICE	ReceiveFuel; GenerateMechanicalPower
Electric Moto-Generator	ReceiveElectricEnergy; GenerateMechanicalPower
Battery	StoreElectricEnergy
Cooling system	MonitorSysHealthStatus
Mechanical transmission	TransferMechanicalPower
Propeller	GenerateTraction

4.3 Design and optimization of hybrid propulsion systems

The second case study concerns the preliminary design of different hybrid versions of a well-known general aviation airplane: the Piper PA-38 Tomahawk. Four design problems are conducted, using the 1st Generation MDO framework described in subsection 3.5.1. The first design problem refers to the preliminary sizing of the conventional Piper PA-38, e.g. the airplane characterized by a traditional (not hybrid) propulsive power. This first application exercise has the main aim of obtaining a known solution, from which deriving design hybrid variants. Furthermore, this case study aims at validating the proposed MDO framework. The second test case shows an application of the proposed methodology for the preliminary design of hybrid-electric propulsion airplanes (subsection 3.3.2). In this design problem a 30% hybridization degree is assumed. The test case shows that the hybridization of the reference aircraft might entail benefits in terms of reductions of gross weight and fuel consumption. The third application example represents a DOE problem. Several hybridization degrees ranging from 0% (i.e. conventional aircraft) to 35% are tested. Resulting designs are compared, defining the best solution, i.e. the one requiring less fuel. Finally, the fourth design problem refers to the application of the Fuzzy Logic for the multi-objective optimization of the hybrid version of the PA-38. Therefore, it can be demonstrated that a “better” aircraft from the customer’s perspective can be obtained through the negotiation of some high level requirements.

Before treating the design of the traditional and hybrid versions of the Piper PA-38, it is below presented the Design Structure Matrix (DSM) set up for this case study (see Figure 40). The DSM is a simplified version of the XDSM, which represents only the connections among the available design disciplines (blocks of the diagonal) without depicting the process flow (e.g. sequence of execution) [4].

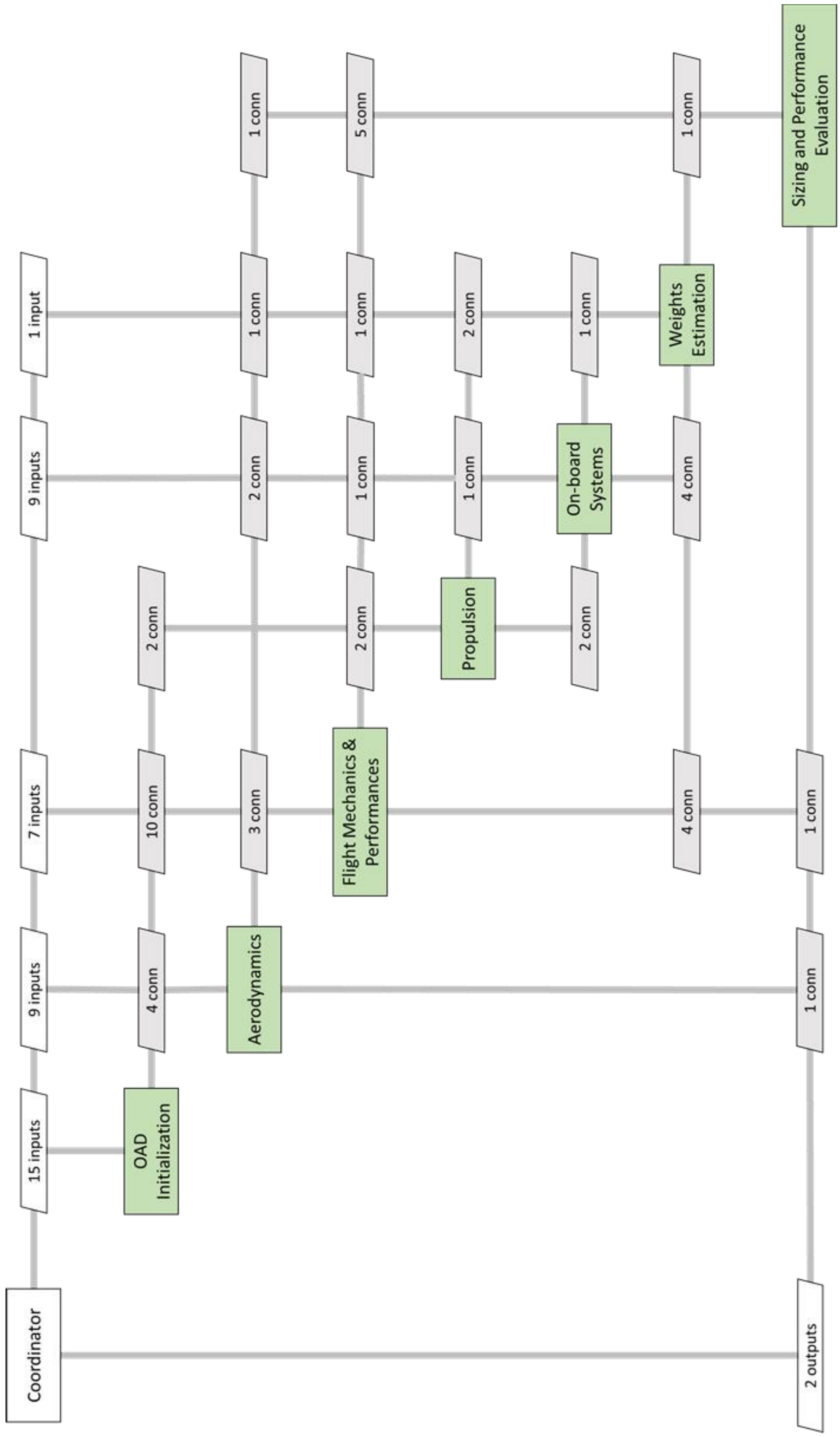


Figure 40: DSM of the 1st Generation MDO.

The diagram is constructed according to the design disciplines described in Section 3.5.1 relative to the 1st Generation MDO. In particular, the coupling variables exchanged among the design modules – e.g. aerodynamics, propulsion and weight estimation – are traced. As example, the on-board system design discipline receives from the aerodynamics discipline the aircraft wingspan for the evaluation of the surface to be de-iced and the mobile surfaces hinge moments for the sizing of the FCS. One connection is present between the propulsion and the subsystems disciplines, i.e. the ICE power. This value is used together with the power needed in take-off (estimated inside the “Performance and Flight Mechanics” module) for the sizing of the electric drive of the hybrid propulsion system. From the weight estimation module instead the on-board systems design discipline receives the MTOW, the OEW, the fuel weight and the MLW. On the contrary, the weight estimation module obtains the total subsystems – included the HEPS – mass, while the shaft and bleed air power off-takes are forwarded to the propulsion discipline.

4.3.1 Reference aircraft

As previously introduced, the Piper PA-38 Tomahawk is selected as reference aircraft in this second case study.

The Piper PA-38 is a two-passenger general aviation airplane powered by a single piston engine (see Figure 41). The main specifications of this airplane are gathered from [192] and listed in Table 11.

Table 11: Piper PA-38 Tomahawk main specifications [192].

Piper PA-38 Tomahawk	
Payload (2 pax) [kg]	163
Empty mass [kg]	512
Fuel mass [kg]	82
Max Take Off Mass [kg]	757
Length [m]	7.04
Wing span [m]	10.36
Wing area [m^2]	11.59
Cruise speed [km/h]	185
Range [km]	870
TO Field length [m]	580
Max engine power [hp]	112

Figure 41: Piper PA-38 Tomahawk [www.commons.wikimedia.org].

4.3.2 Case 1: Design of the conventional Piper PA-38

The first design case relates to the conventional Piper PA-38 Tomahawk, meant as the airplane characterized by the traditional propulsion system. The main aim of this first case study is the calibration of the equations and algorithms of the proposed methodology in order to complete a conceptual design of a “real” reference aircraft whose main specifications are publicly available. The obtained results will represent the starting point of the following case studies. In other words, new hybrid versions will be derived by modifying the resulting traditional aircraft.

Other than calibrating the design equations and the algorithms, the present case study is employed to validate the 1st Generation MDO framework proposed in the dissertation, at least for the development of general aviation airplanes. Moreover, it must be noted that the proposed design system encompasses design methodologies already validated in literature.

This kind of design problem is a converged MDA, as represented by the XDSM of Figure 42. Differently from the DSM previously described, the diagram highlights the order of execution of the blocks on the diagonal. In particular, the block “Converger” is introduced, identifying the convergence on the MTOW, keeping the OAD Initialization module outside the convergence loop.

The main results of the design methodology are collected in Table 12.

Table 12: Results comparison between the “real” Piper PA-38 and the “designed” Piper PA-38.

	“Real” Piper PA-38	**“Designed” Piper PA-38**	**Difference**
Empty mass	512 kg	512 kg	0%
Fuel mass	82 kg	82 kg	0%
Max Take-Off Mass	757 kg	757 kg	0%
Wing span	10.36 m	10.39 m	-0.3%
Wing area	11.59 m^2	11.66 m^2	-0.6%
Max engine power	112 hp	112 hp	0%

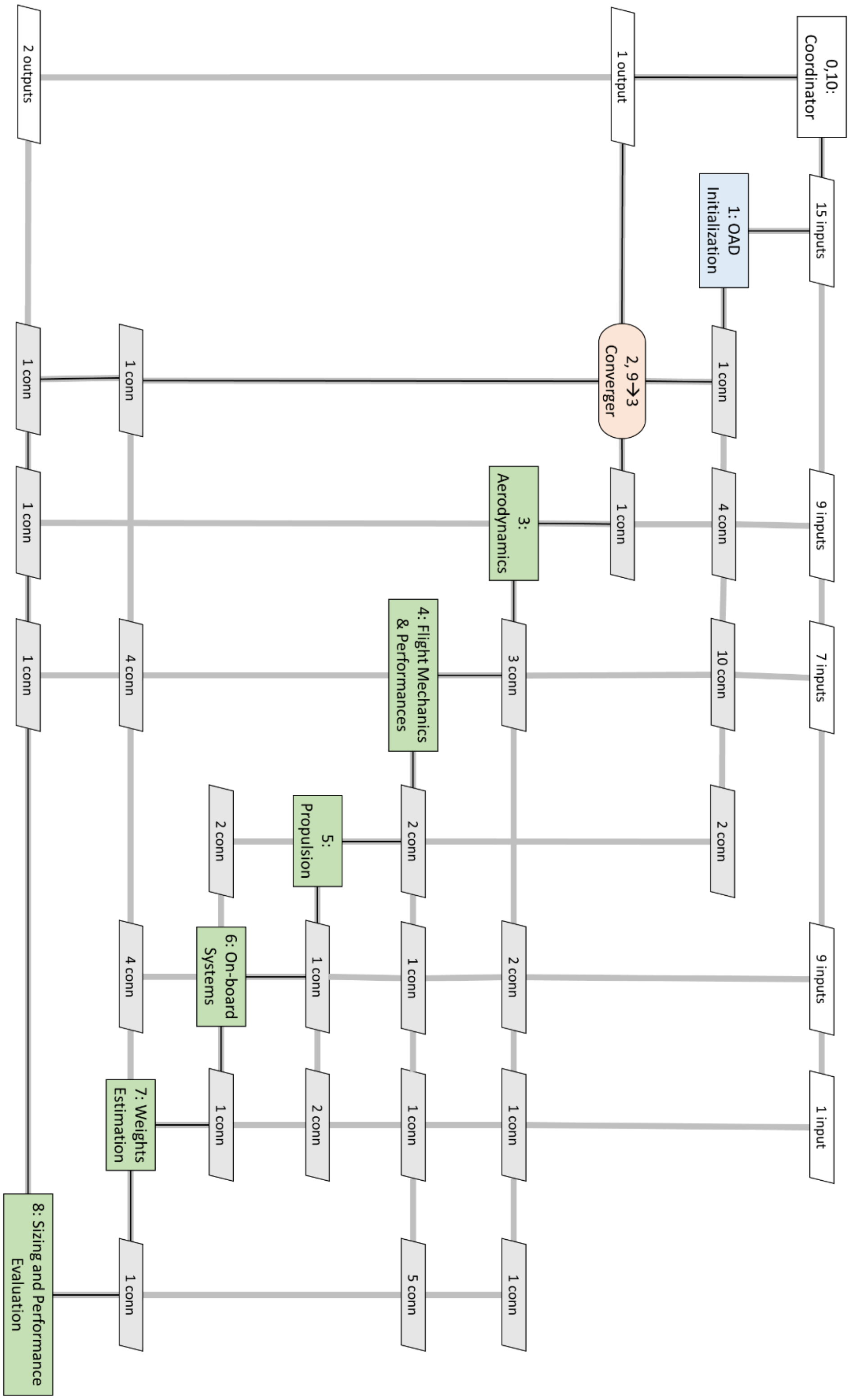


Figure 42: XDSM of the 1st Generation MDO (converged MDA problem).

It can be noted that at the end of several iterations, almost equal resulting values are found. The exceptions are the dimensions of the wings, even if the difference between the “real” and the “designed” aircraft is minimal (less than 1%).

Other main results of the design are shown in Figure 43 and Figure 44. Figure 43 represents the “Power Loading vs. Wing Loading” diagram of the aircraft. Three propulsive power requirements are considered, specifically the take-off, the cruise and the climb requirements. The two last requirement curves are almost overlapping, while during take-off a higher amount of power is needed. Hence, a little gap of power loading between cruise (or climb) and take-off exists. The next case study will be devoted to the minimization of this gap by designing a HEPS. Specifically, the thermal engine will be optimized for the climb and cruise conditions, while the hybrid drive will be designed in accordance with the take-off requirements.

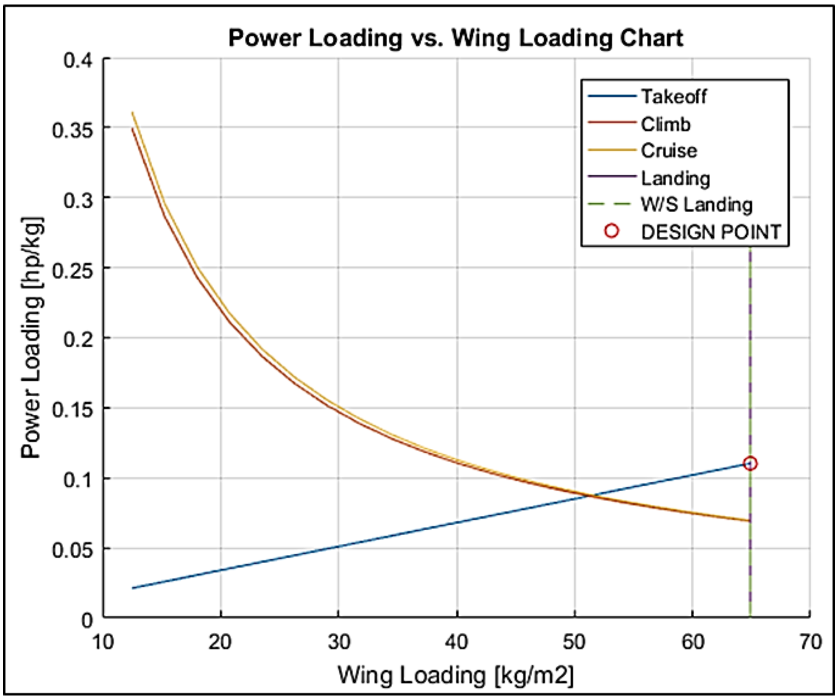


Figure 43: Graphic results of the conventional Piper PA-38: Power Loading vs. Wing Loading Chart [143].

The climb profile is illustrated in Figure 44, showing the variation of altitude both in space and time. The climb profile is evaluated considering the throttle of the piston engine set at continuous power. It is worth noting that decreasing the level of climb power implies an extension of the climb profile. In other words, the duration of the climb phase is increased by a downsize of the thermal engine. This fact is evident in the case of the design of hybrid airplanes, as will be demonstrated in the following subsections.

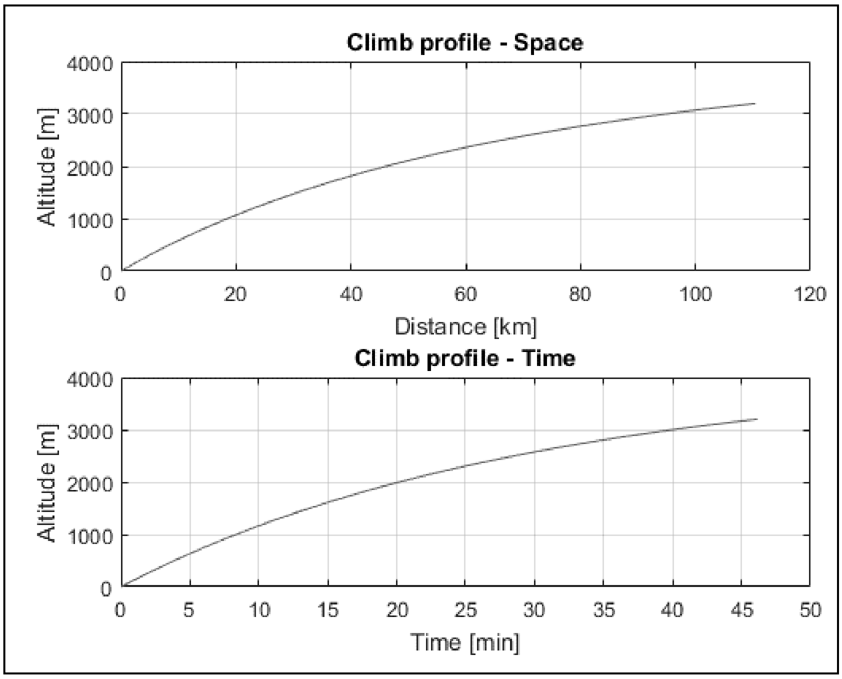


Figure 44: Graphic results of the conventional Piper PA-38: Climb profile [143].

4.3.3 Case 2: Design of a 30% hybrid Piper PA-38

After having re-designed the traditional Piper PA-38, the proposed methodology is adopted to develop a hybrid version of the same airplane. In particular, the same high level requirements are considered, while additional requirements are included. In this first hybrid case, the hybridization degree is set to 30%, entailing changes in propulsion system and aircraft masses.

Concerning the electric taxi phase, a 2000 m long asphalt taxiway (μ=0.02) is hypothesized connecting the parking of the airplane and the take-off runway. In this phase, the general aviation aircraft is supposed to move with a ground speed of 8.5 m/s, hence requiring a nearly 4 minutes long taxi. From these inputs, having selected a variable pitch propeller with a diameter of 1.83 m, an angular speed of 2200 RPM and a propeller efficiency equal to 0.5, the blades traction and the shaft power are assessed. From these results, a mechanical power of 1.47 kW is requested to the electric motor, considering a propeller-EM transmission efficiency of 0.98.

The original 112 hp thermal engine is replaced with a smaller and lighter 74 hp ICE. The mass of the new engine is scaled down by a linear relationship, deducting the masses of the starter and the electric generator. The new engine resulting mass is equal to 69 kg, while the original ICE weights 112.5 kg [193]. As shown in the "Power Loading vs. Wing Loading" diagram of Figure 45, the new selected engine is suitable for the cruise requirement, while it appears to be undersized for the take-off condition. In this phase, indeed, the gap of power needed for the compliance with the take-off requirement is covered by the electric motor. For this reason, an amount of 23.1 kW of power should be generated by the electric drive, while the piston engine provides all the available power. From the design it appears that the taxi electric power requirement is less demanding than the take-off one. Thus, the electric motor is supposed to guarantee a maximum power of 23.1 kW. Since this value and considering a power-to-mass ratio equal to 2 kW/kg, an electric motor weighting nearly 12 kg is selected, comprehensive of the power controller and installation.

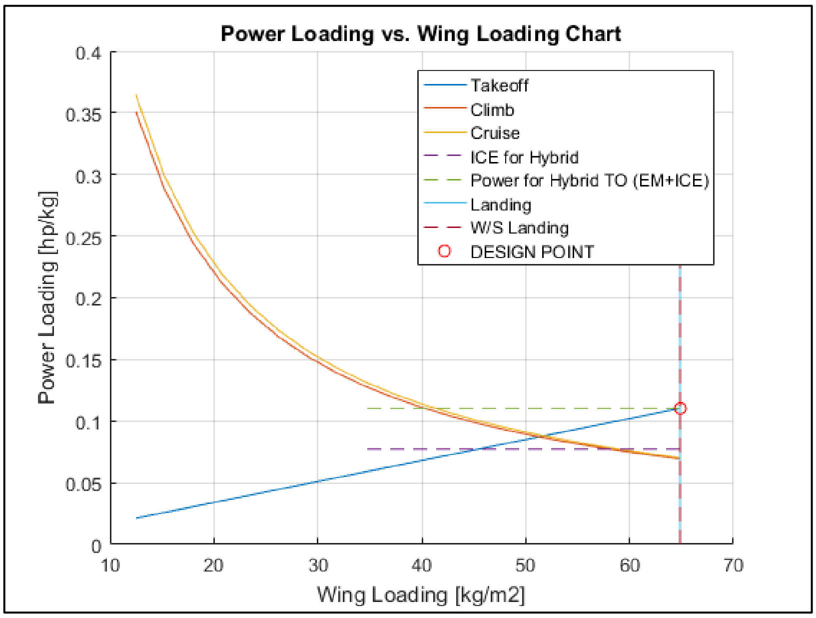


Figure 45: "Power Loading vs. Wing Loading" Chart of the 30% hybrid Piper PA-38 [143].

As explained in subsection 3.3.2, the hybrid propulsion system requires an energy storage system to provide electrical power to the electric machine and to the users when the thermal engine is switched off (during taxi and in case of engine failure) and during the take-off. A 1403 Wh Lithium Polymer battery with an energy density of 140 Wh/kg and a discharge efficiency of 85% is installed, with a total mass of 10 kg.

Finally, the mass of the mechanical transmission is estimated to be equal to 3 kg, while a miscellanea (e.g. installation, propeller, and cooling system) mass is considered unvaried respect to the reference Piper aircraft and hence it is not considered in the mass breakdown.

The main specifications of all the components installed inside the HEPS are summarized in Table 13. As explained previously (subsection 3.3.2), the propulsion system affects the entire aircraft and its performance. First, the OEM of the airplane is reduced from 512 kg to 432 kg. Then, thanks to a lower empty mass and a minor fuel consumption in cruise – due to the thermal engine working in a higher efficiency operating condition – and null during taxi, the fuel mass is lowered to 72 kg. Maintaining the same payload of the original Piper airplane, the new MTOM becomes equal to 714 kg.

Table 13: Resulting specifications of the 30% hybrid Piper PA-38 (adapted from [143]).

ICE mass [kg]	69
Mechanical transmission mass [kg]	3
Electric Motor mass [kg]	12
Battery mass [kg]	10
Total Hybrid Propulsion System mass [kg]	**94**
Empty mass [kg]	483
Fuel mass [kg]	72
Maximum Take-Off Mass [kg]	**714**

In Table 14 are listed additional results of the case study, in particular regarding the altitude gained and lost, the distance flown and the electrical energy used from the taxi phase to the emergency descent.

Table 14: Additional results of the 30% hybrid Piper PA-38 (adapted from [143]).

Phase	**Delta Altitude ΔH [m]**	**Delta Distance ΔX [m]**	**ΔX/ ΔH**	**Battery Energy [Wh]**
Taxi	0	2000	-	567
Take-off	15	349	-23.3	298
Climb	113	1315	-13.4	0
Crosswind Leg	-40	577	14.4	103
Downwind Leg	-33	1664	50.4	332
Base Leg	-40	577	14.4	103
TOTAL	-	6483	-	1403

An installed propeller efficiency of 0.7 is assumed for all the above mentioned flight segments, except for the taxi, during which the efficiency drops to 0.5.

4.3.4 Case 3: DOE of hybridization degrees

Eight architectures characterized by different hybridization degrees are designed. The XDSM referred to the present design problem is schematized in Figure 46.

In the first design case, a conventional propulsive system is installed, while in the other designs the hybridization degree ranges between 5% and 35%. Hybridization degrees over 35% are unfeasible without reducing the flight altitude or the cruise speed, as the propulsive power is not sufficient to balance the drag force, avoiding a rectilinear flight. The purpose of this analysis is to compare the different solutions in terms of power, masses and minimum safety altitude, eventually defining the optimal one. The optimal design solution in this dissertation is the one with the lowest fuel mass – and hence MTOM – and characterized by the minimal safety altitude. All the main obtained results are collected in Table 15.

Table 15: Results comparison of different hybridization degrees (adapted from [143]).

	Hybridization degree							
	0%	**5%**	**10%**	**15%**	**20%**	**25%**	**30%**	**35%**
ICE power [hp]	112	113	104	96	89	81	74	68
EM power [kW]	0	4.3	8.4	12.3	16.1	19.7	23.1	26.5
Minimum safety altitude [m]	220	415	298	313	193	157	128	105
Empty mass [kg]	512	560	536	521	510	494	483	471
Fuel mass [kg]	82	84	81	78	76	74	72	70
Max. Take-Off Mass [kg]	757	807	780	762	749	731	714	704

It can be noted (Figure 47 (a)) that the hybrid propulsion brings to an increase of the aircraft empty mass for low values of hybridization degree. This fact is due to the introduction of the gearbox, the heavier electric motor and the installation of the energy storage system. The downsize of the piston engine is not sufficient to compensate the mass increment of the propulsion system. However, for higher hybridization degrees, the OEM decreases, partly due to the lighter thermal engine, but mainly because of the reduction of the SFC, which entails strong fuel savings, as shown in Figure 47 (b). The reduction of the empty and fuel mass impacts the MTOM. In Figure 47 (c), it can be seen that the take-off mass is lower than the conventional configuration for hybridization degrees over 15%. Furthermore, the "snowball" effect is captured in the design, i.e. the reduction of the empty mass for

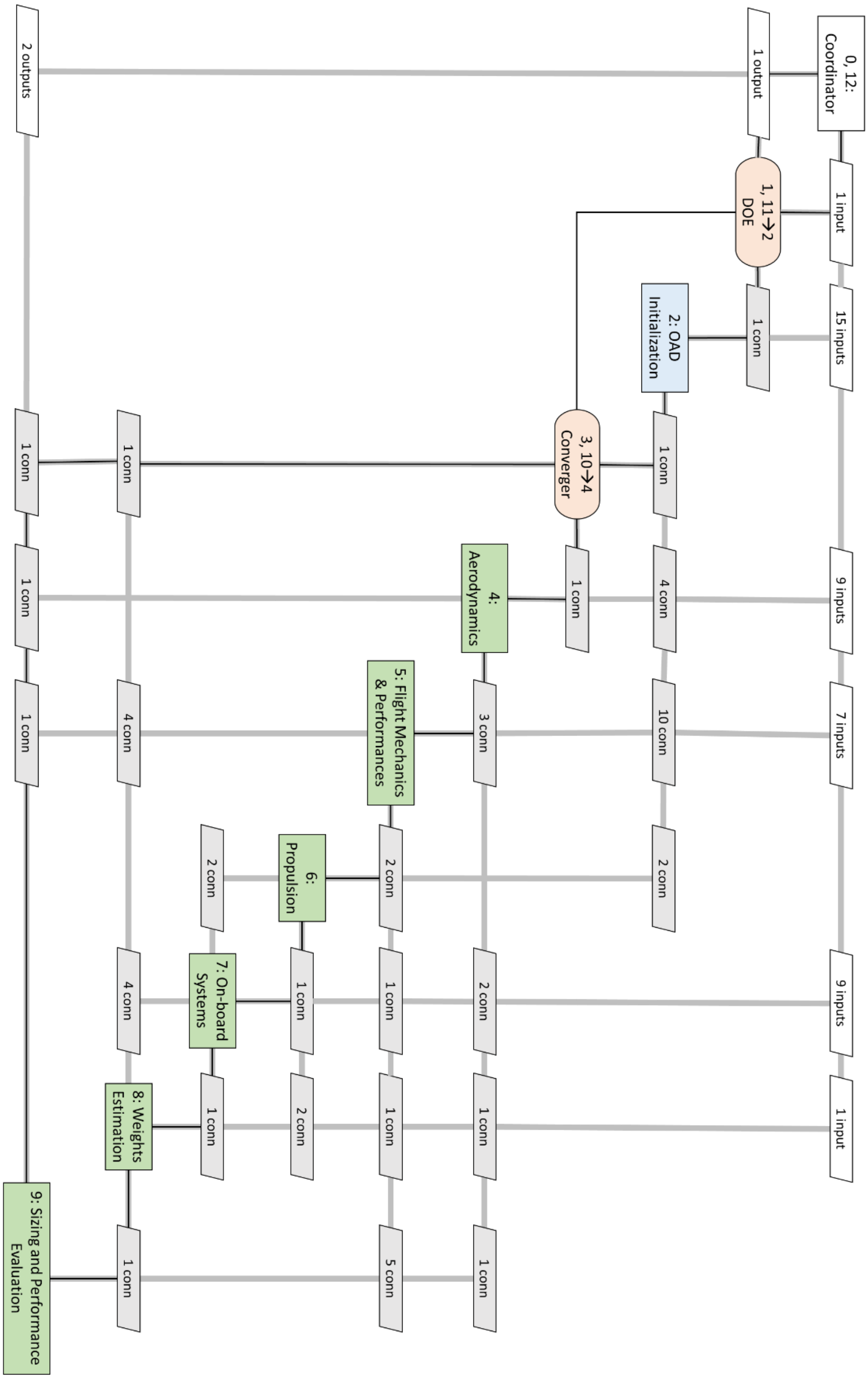


Figure 46: XDSM of the 1st Generation MDO (DOE converged MDA problem).

high hybridization degrees is due to the high decrease of the fuel mass. Vice-versa, the fuel mass reduces not only because the more efficient thermal engine, but also due to the decrease of the MTOM, which is linked with the airplane empty mass.

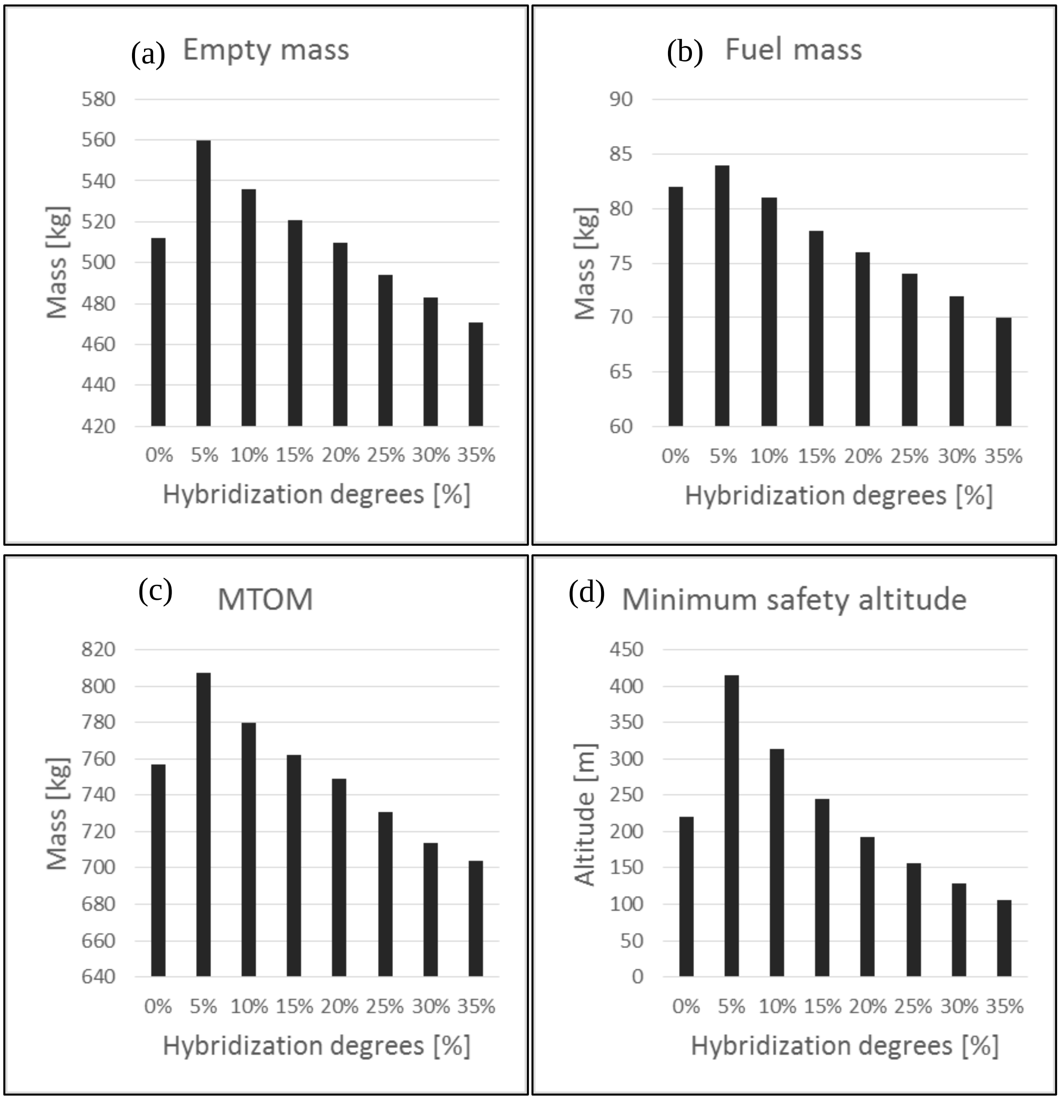


Figure 47: Bar charts of the design results varying the hybridization degree: (a) Empty mass; (b) Fuel mass; (c) MTOM; (d) Minimum safety altitude [143].

The last bar chart of Figure 47 compares the minimum safety altitude of the different eight configurations. It is worth noting that for higher hybridization degrees the minimum safety altitude drastically decreases, therefore raising the safety level. Two reasons are at the base of this altitude reduction. The first one is the increment of the propulsive (electric) power during the emergency descent, which entails a consistent reduction of the descent angle. The second reason is the reduction of the MTOM. As a lighter airplane requires less space to perform the return maneuver, the minimum altitude at which an engine failure could occur without dramatically affecting the safety of the aircraft is lower.

From these outcomes, the 35% hybrid configuration results the optimal one, because it is characterized by the lowest fuel and maximum take-off masses and by the lowest minimum safety altitude. However, the reduction of the thermal engine brings to a worsening of some flight performance characteristics. For instance, the diagrams in Figure 48 show that for higher hybridization degrees the climb phase increases in time and space.

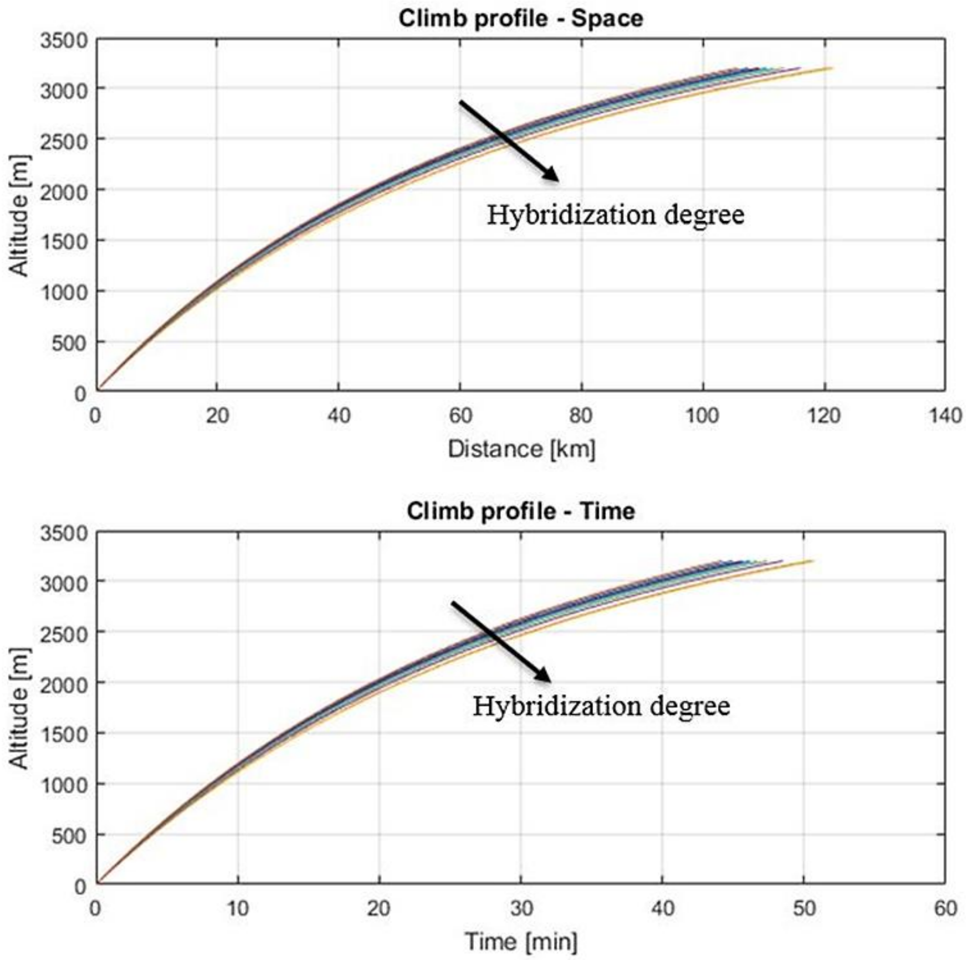


Figure 48: Climb profiles for different hybridization degrees [143].

4.3.5 Case 4: Multi-objective optimization through the Fuzzy Logic

The last design problem conducted on the Piper PA-38 is an application of the Fuzzy Logic (see subsection 3.4.1). The main aim of this study is the negotiation of some high level requirements for the determination of a better (i.e. optimal) aircraft, from the customer's perspective. Therefore, this fourth design problem is a multi-objective optimization problem. Four objectives are considered in the present case study: 1) minimization of the fuel mass; 2) minimization of the MTOM; 3) minimization of the duration of the climb phase, and 4) minimization of the minimum safety altitude (hence maximizing the difference with the conventional aircraft). Some high level requirements are considered as design parameters that could be negotiated to improve the design as perceived by the customer. These parameters are: 1) the flight range; 2) the payload mass; 3) the TOFL, and 4) the cruise speed. In the proposed case study, these four parameters are considered also as design objectives, increasing the number of objective functions up to eight. Moreover, an additional design parameter is included: the hybridization degree. However, differently to the other parameters, the value of the hybridization degree is not a design objective, but it contributes to achieve a better and more optimized, aircraft.

The XDSM diagram reported in Figure 49 represents this case study. It can be noted that a block is added at the end of the flow process, namely the "Fuzzy Scores Assignment" block. This module evaluates the global fuzzy score, which is returned to the "Optimizer" block.

The objective functions and relative fuzzy sets considered in the current MDO problem are listed in Table 16.

Table 16: Objective functions and relative fuzzy sets [148].

	***ymin* (*μ*=1)**	***ymax* (*μ*=0)**
MTOM [kg]	600 kg	800 kg
Fuel mass[kg]	65 kg	85 kg
Climb duration [min]	20 min	60 min
Safety altitude [m]	50 m	250 m
Range [km]	870 km	710 km
Payload [kg]	163 kg	150 kg
TOFL [m]	580 m	630 m
Flight speed [km/h]	185 km/h	150 km/h

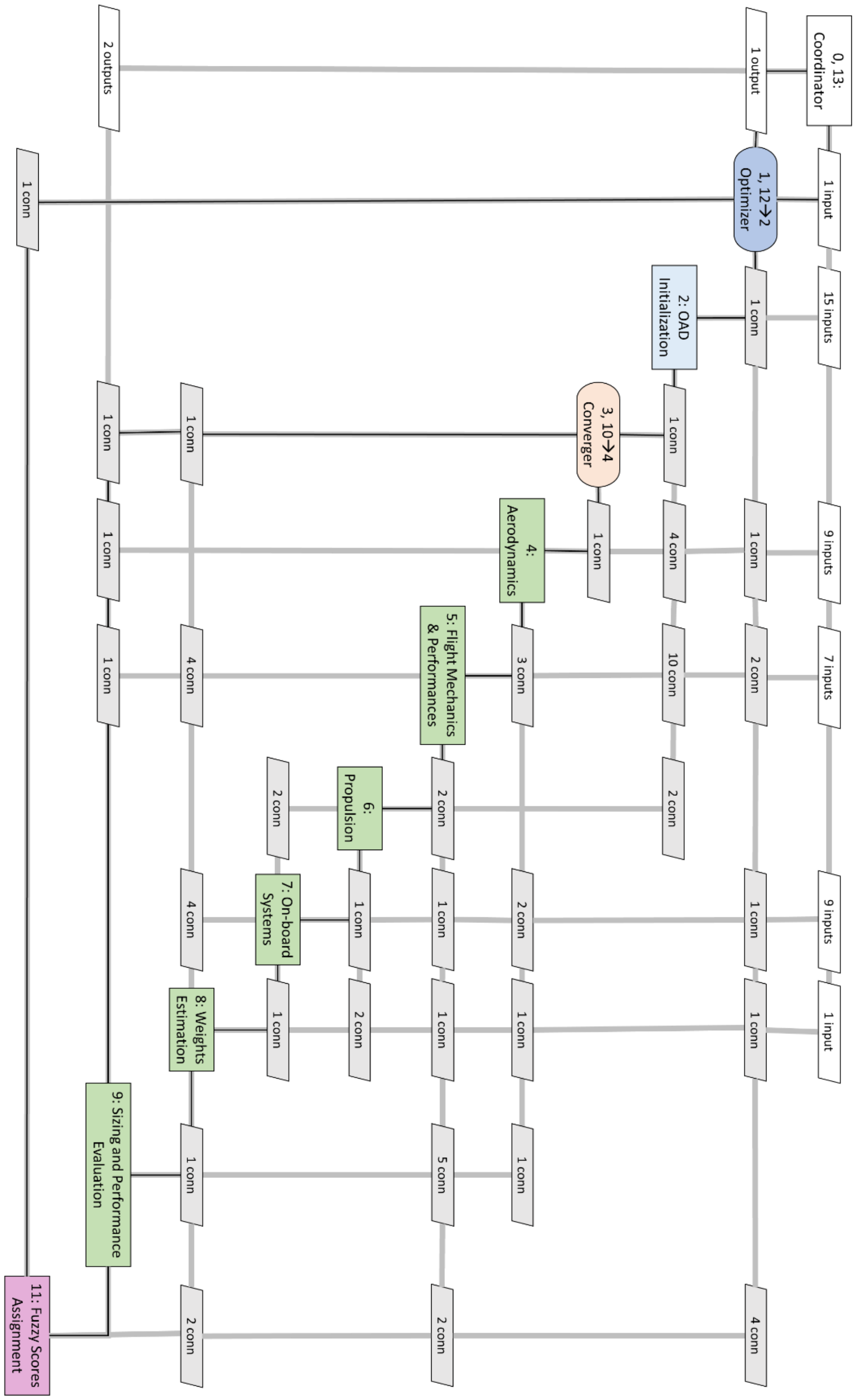


Figure 49: XDSM of the 1st Generation MDO (optimization problem).

The additional design variable – the hybridization degree – could assume a value equal to 0% for the case of the conventional (i.e. non-hybrid) aircraft, or it could be greater than zero with values up to 40% for hybrid vehicles. However, no fuzzy set is associated to the hybridization degree, though it contributes to the determination of the optimal solution.

The optimization process starts selecting as initial design vector the un-relaxed values of the high level requirements and the hybridization degree equal to 0%. This solution corresponds to the conventional airplane, whose resulting specifications are collected in the second column of Table 17. For this solution, a satisfaction degree of 0.15 is calculated. Therefore, the optimizer finds the optimal solution characterized by the highest global fuzzy score. The results of the optimal solution are reported in the third column of Table 17, in which are also gathered the main negotiated requirements, namely the TOFL, the flight speed, the payload and the range. As a consequence, some resulting design objectives are improved with respect to the conventional solution. As example, the fuel mass is reduced from 82 kg to 62 kg. On the contrary, other design objectives are worsen. Other than the relaxed high level requirements, which are also considered objective functions, the climb duration is increased of nearly 50%. Moreover, in the optimal solution a hybrid propulsion system with a hybridization degree of about 30% is installed. Thus, the customer’s satisfaction degree of the new solution is over three times higher than the case of conventional baseline, meaning that the hybrid solution is more appreciated by the stakeholders.

Table 17: Results of the multi-objective optimization [148].

Specifications	**Piper PA-38 (conventional)**	**Piper PA-38 (hybrid-optimized)**
Hybridization degree [%]	0	30,82
TOFL [m]	580	605.24
Flight speed [km/h]	185	185
Payload [kg]	163	156.46
Range [km]	870	781.6
MTOM [kg]	757	670.32
Fuel mass [kg]	82	65
Climb duration [min]	21	30.34
Safety altitude [m]	205	87.6
Satisfaction degree	0.15	0.4951

An additional study is then performed starting from the obtained optimal solution. The design space in proximity of the optimal solution is investigated. These design parameters are the hybridization degree, the TOFL, the payload and the range. This investigation is performed with a DOE consisting of about 30000 aircraft designs characterized by several combinations of the design parameters. For each designed solution, a global fuzzy score is calculated. With the aim of visualizing the variation of the satisfaction degree within the design space, the contour lines obtained through each experiment are sketched. The contour lines represent the aircraft designs characterized by constant degrees of satisfaction. The color of these lines is affected by the values of the global degree. Lighter colors state best solutions, while darker colors represent lower global fuzzy scores. In Figure 50 are reported the contour lines of the DOE, in which the TOFL and the range are considered constant to those relative to the optimal solution (the ‘X’ mark indicates the optimal solution). In particular, the plot represents a portion of the DOE problem, which contains only one third of all the solutions. The range and TOFL values are kept constant, while the payload is let varying from 156 kg to 157 kg and the hybridization degree ranges from 30% to 31%. It can be seen from Figure 50 that the global fuzzy score decreases for high hybridization degrees (i.e. over than 30.8%). High hybridization degrees entail reductions of the propulsive power of the thermal engine. Due to this downsize, the airplane has not enough propulsive power to maintain the steady flight at the cruise altitude with the required speed. Therefore, the flight speed is then decreased to limit the required propulsive power, hence negatively affecting the satisfaction degree. Moreover, it is worth noting the different orientation of the contour lines. For values of payload lower than 156 kg, the lines are almost vertical. In fact, in this design area, the fuzzy scores of the payload are dominant, i.e. they represent the minimum of eq. 21. On the right side of the diagram instead, the payload assumes values low different from the requirement. Thus, the contour lines are sloping, as all the four design objectives are affecting the global fuzzy score. The variation of the color of the contour lines reveals the portion of the design space where the optimal solution is located. It can also be noted that a very narrow range of payload and hybridization degree is depicted in Figure 50. A broader area of the design space might be indeed investigated through the proposed method. However, the colors of the lines sketched in the diagram show the different slopes of the satisfaction degrees around the optimal solution along all the directions. The optimal solution is indeed placed in proximity of a high sloping area. Increasing the hybridization degree maintaining constant the payload, the range and the TOFL would bring to an excessive reduction of the flight speed, entailing an unacceptable solution for the customer. Analogously, further reductions of the payload would cause solutions with extremely low satisfaction degrees. Small reductions of the hybridization degree and small increments of the payload instead would bring to solutions characterized by a high satisfaction degree.

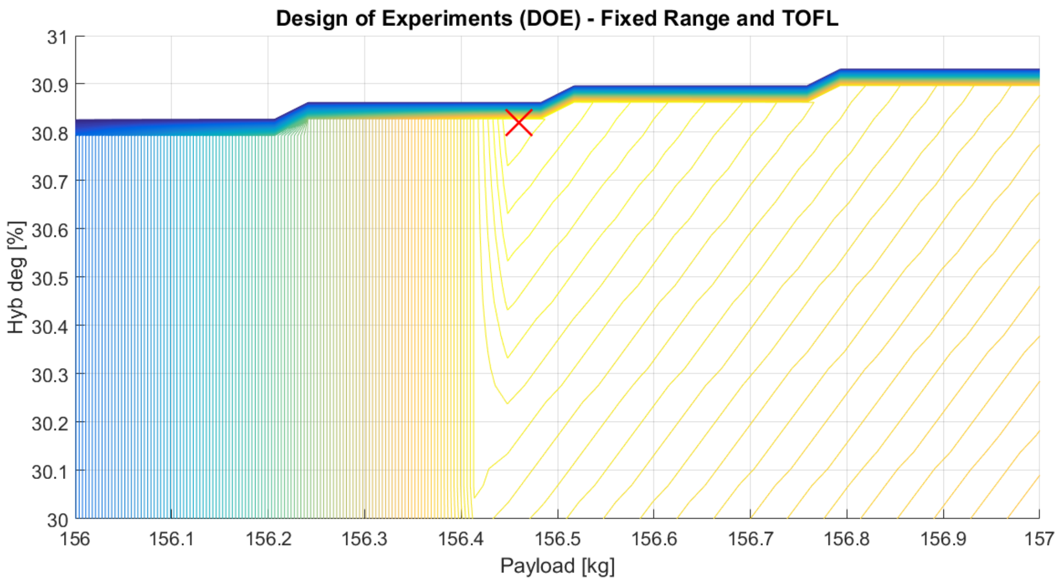


Figure 50: Contour lines of the global fuzzy score – Fixed Range and TOFL [148].

4.4 Design and optimization of on-board system architectures

The methodology proposed in the current dissertation is employed for the design and optimization of several on-board system architectures. Both conventional and innovative subsystem configurations – as those briefly described in subsection 1.2.1 – are considered.

A 90-passenger regional jet is selected as reference aircraft, and several levels of subsystem technology are considered, identifying the various on-board system architectures. From this set of requirements, a first conceptual design of the entire aircraft is performed within the AGILE project. The results are used for the validation and calibration of the 1st Generation MDO framework described in subsection 3.5.1. Then, four design studies are conducted, considering four types of subsystem architectures, from conventional to All Electric. Particular attention is posed on subsystems discipline, especially concerning the estimation of the masses and the assessment of the power off-take budgets. These studies are done initially employing the proposed 1st Generation MDO framework. A more detailed analysis is also performed by means of the AGILE's distributed 3rd Generation MDO framework, in which the disciplines are provided by various experts of the AGILE consortium [194]. The results obtained by both the MDO frameworks are in line with the studies present in literature, evidencing the possible benefits of the All Electric architecture ([97], [189], [96], [135]). Finally, a DOE of 512 system architectures is presented, each one characterized by different technological choices, for instance typology of actuators, types of electric generators, voltages, bleed/bleedless configuration.

4.4.1 Reference aircraft

In the third presented case study, the high level requirements of the reference aircraft are derived from the AGILE project and listed in Table 18 [194]. These TLARs have been defined by Bombardier. The chosen reference aircraft consists of a conventional 90-passenger transport liner, characterized by a traditional "tube-and-wings" configuration, with two jet engines mounted under the wings. The choice of a conventional reference airplane is led by the objective of testing and validating the technologies developed within the context of the AGILE project. Bombardier has eventually been in charge of approving the results of the design problem.

Table 18: Requirements of the civil regional jet [194].

Civil regional jet	
Range	3500 km
Passengers	90 @ 102 kg
Max Landing Weight (%MTOW)	90%
Max Operating Altitude	12500 m
Residual Climb Rate	90 m/min
TOFL (ISA, Sea Level, MTOW)	1500 m
Fuselage diameter	3 m
Fuselage length	34 m
Fuel reserves	5%

From this set of TLARs, a first not optimized design solution is determined by the AGILE consortium. The main aircraft design results are collected in Table 19. These results are obtained after several design iterations, as reported in [195]. It can be noted that the resulting solution is close to similar airplane of the same class and with analogous mission profile, as the Embraer E170.

Table 19: Main results of the AGILE reference regional jet [195].

Aircraft masses	
MTOM [kg]	45000
OEM [kg]	27000
Max fuel mass [kg]	8100
Main wing geometrical data	
Wing area [m^2]	75
Wing ¼ chord sweep [deg]	26.2
Wing Aspect Ratio [-]	9.5
Wing Taper Ratio [-]	0.217
Main propulsion system data	
Max engine net thrust [kN]	90
Engine clean SFC [lb/lb/h]	0.575

A preliminary 3D model of the regional jet under development is derived, as shown in Figure 51. The obtained geometries, together with all the other results, define a baseline for the further preliminary design of the on-board systems. In particular, in the current dissertation four types of subsystems are considered, as described in the following part of the current Section.

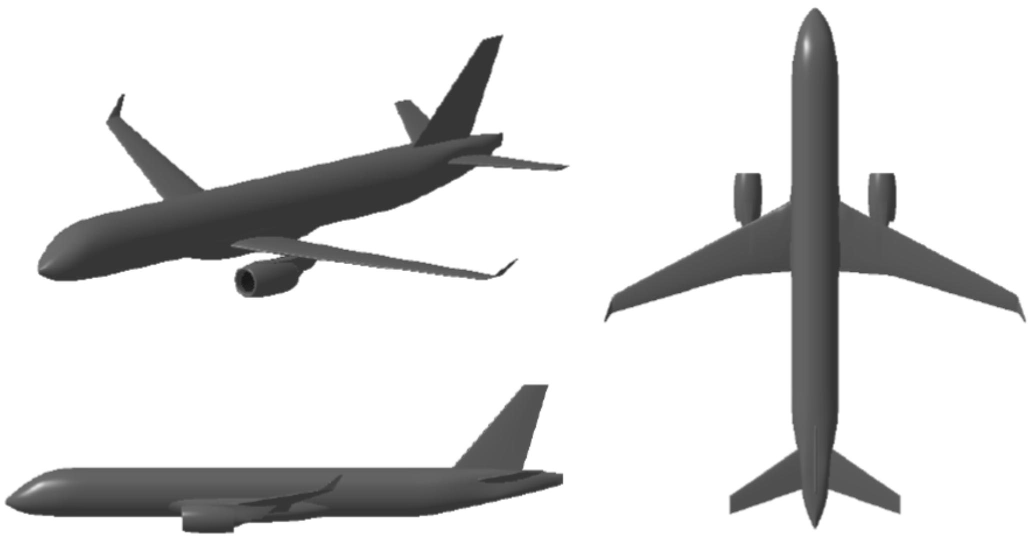


Figure 51: 3D model of the reference regional jet [195].

Four on-board system architectures

The four on-board system architectures considered in this case study are graphically schematized in Figure 52.

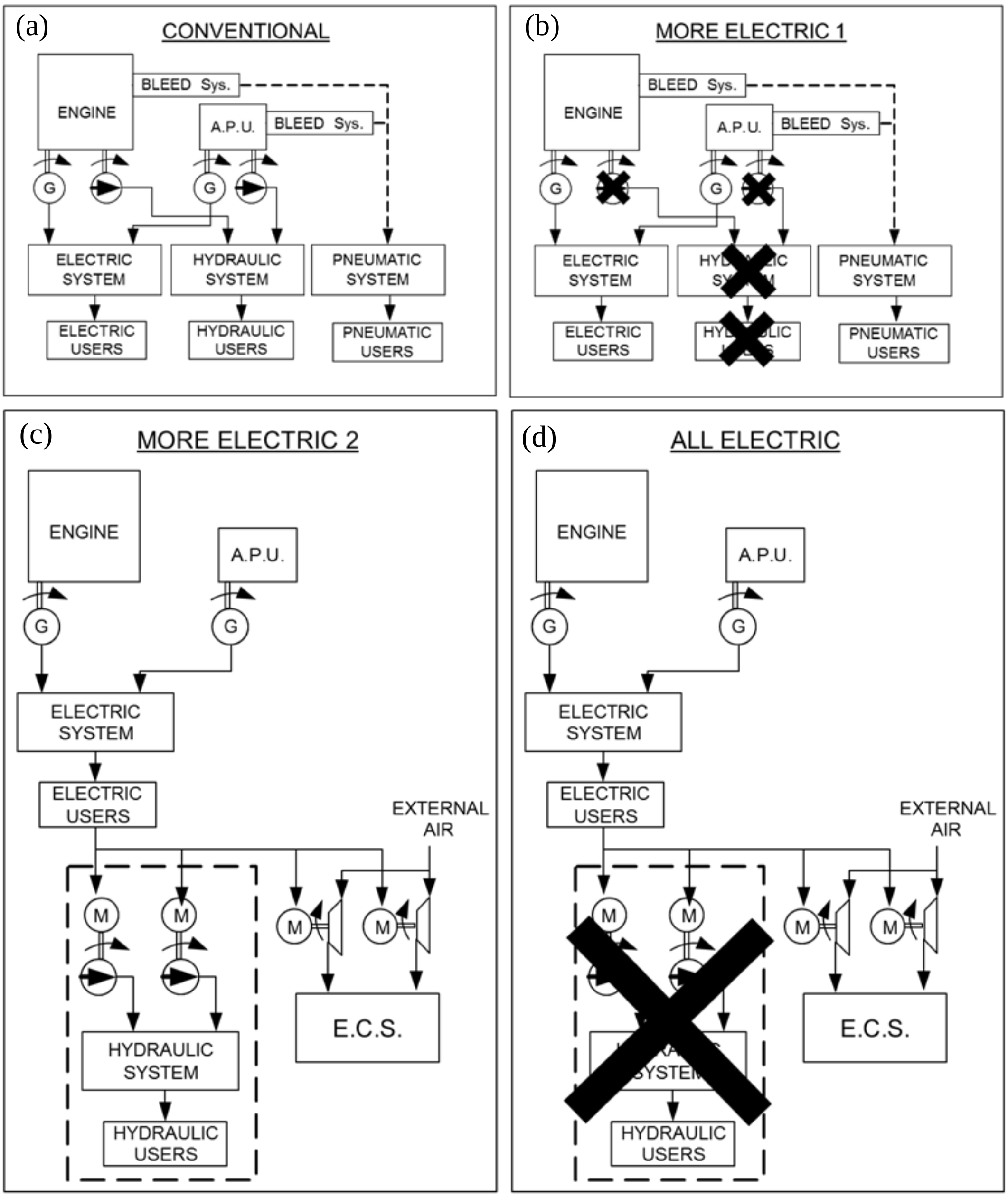


Figure 52: Four on-board system architectures [175].

The technological level of these configurations ranges from conventional to All Electric. The main differences among traditional and innovative architectures and the most important advantages and potentialities introduced by newer configurations have been presented in subsection 1.2.1. Hereunder are briefly described the conventional and the three subsystem innovative configurations, in particular highlighting the peculiar innovations of the unconventional solutions.

The first investigated architecture is the conventional one, hereafter denoted as “CONV” (Figure 52 (a)). This solution is characterized by the generation of all the three typologies of non-propulsive power: electric, hydraulic and pneumatic. Mechanical shaft power off-takes are extracted by AGBs from the engines and the APU. This mechanical power is then converted by Integrated Drive Generators (IDGs) in electric 115 V AC power at constant frequency (400 Hz). Transformer Rectifier Units (TRUs) shall be sized for the transformation of electric power in 28 V DC. Analogously, part of the mechanical power is employed for the alimentation of engine-driven hydraulic pumps, which pressurize the hydraulic oil up to nearly 20.7 MPa (3000 psi). The generated electricity is employed to supply power to avionics, lights, IFE, fuel pumps, miscellanea (e.g. ovens, chillers, small users ice protection system). The hydraulic power instead is demanded by the actuators of moveable surfaces and landing gears, and by the braking system. Pneumatic users, as the WIPS, the CIPS and the ECS are supplied by hot and high pressure air bled from the engine compressors and from the APU.

The first innovative on-board system architecture is represented in Figure 52 (b). It is named “MEA1” (from More Electric Aircraft), as it is one of the two more electric configurations considered in this study. This solution is characterized by the removal of the hydraulic system. Therefore, all the FCS and landing gear actuation systems are supplied by an augmented high voltage EPGDS. A permanent magnet alternator coupled with an electronic converter is installed on each engine and on the APU, generating 235 V AC at variable frequency. This electric power is later transformed in 115 V AC or rectified to 270 V DC and 28 V DC. Furthermore, part of the total power generated by the propulsion system is in the form of pneumatic high pressure airflow, as the conventional case.

On the contrary, the peculiarity of the second more electric architecture (“MEA2”) is the generation of only electric secondary power, as depicted in Figure 52 (c). This solution is a “bleedless” configuration, as the bleed air system is removed. Thus, the wing de-icing system and the air conditioning system are supplied by the electric power. In the first case, an innovative WIPS based on electric resistors is sized. Concerning the ECS, electrically driven air compressors are designed. The removal of the pneumatic system entails two consequences. First, the total demand of electric power increases considerably. Therefore, high voltage electric current shall be generated, as the previous case. This fact entails also an increment of the size of the generators. Furthermore, the jet engine cannot be started pneumatically, but electrically. Hence, electric starter-generators are installed. Finally, the HPGDS is still present, but the hydraulic oil pressure is raised up to about 34.5 MPa (5000 psi). Furthermore, this solution is characterized by more efficient electric driven hydraulic pumps, instead of engine driven pumps.

The last solution – named “AEA” (All Electric Aircraft) – is the most innovative all electric architecture here considered (Figure 52 (d)). It combines the peculiarities of the two more electric architectures. The electric power is the only one generated with this innovative configuration. All the actuators, the braking system, the WIPS and the air conditioning system are supplied by a high voltage electric system.

4.4.2 Design of the aircraft on-board systems

All the on-board systems of the previously described architectures are designed by means of ASTRID considering as baseline the AGILE project reference aircraft. Among all the aircraft subsystems, attention in this Section is focused on FCS, landing gear, WIPS, ECS, hydraulic, pneumatic and electric systems, as their design varies according to the technological level.

The design of the FCS starts with the definition of the subsystem architecture, outlining the number of movable surfaces, actuation speed and stall moment or force. Therefore in this case study the following movable surfaces are included, as reported in Table 20: two ailerons, two elevators, one rudder, four speed-brakes, six ground spoilers, four leading edge slats and four trailing edge flaps. Two linear actuators are considered for each primary surface, while a single actuator is required by each spoiler and speed-brake. In case of conventional architecture, these actuators are hydraulically driven by a centralized hydraulic system. Otherwise, newer architectures are characterized by innovative EHAs, supplied by the EPGDS.

Regarding the landing gear design, a tricycle arrangement is considered. In particular, all the three landing gear struts are retractable, while the braking system is installed on the main gear and the steering system concerns the nose landing gear. As described in subsection 3.3.2, the retraction power depends on the mass of the struts, on the distance between the leg hinge and the strut center of gravity (refer to Figure 19), and by the retraction speed. All these parameters and the resulting values are collected in Table 21.

Table 20: Main specifications of the regional jet movable surfaces ([196], [197], [198], [199].

Movable surface	Actuation speed	Stall moment/force
Ailerons	60°/s	4200 Nm
Elevators	60°/s	7600 Nm
Rudder	60°/s	8200 Nm
Speed-brakes	60°/s	4200 Nm
Ground spoilers	40°/s	3800 Nm
Leading edge slats	102 mm/s	51 kN
Trailing edge flaps	60 mm/s	6.3 kN

Table 21: Main specifications of the regional jet landing gear retraction system.

Strut	Strut mass [kg]	Distance hinge-CoG [m]	Retraction speed [°/s]	Retraction moment [kNm]
Nose gear	~ 135	1.2	11	1.6
Main gear – left	~ 610	1.2	11	8.9
Main gear – right	~ 610	1.5	11	8.9

For the estimation of the steering power, a static load $L_N = 81384$ N is obtained by means of eq. 2. Thus, assuming a wheel radius of 0.3 m, the steering moment results equal to 8.5 kNm.

Regarding the braking system, the brakes are supposed to contribute to the generation of 50% of the total force decelerating the aircraft during landing. Four disks are installed on each one of the four braked wheels, entailing a total braking power of 1.3 kW.

Turning to the wings de-ice system, the first step refers to the estimation of the total protected area. Part of the wings should be protected from the ice accretion, while the horizontal and vertical tails are supposed to be not protected. A span-wise length of about 55% of the wing span is hence considered. Therefore, the total protected area of the wing leading edges results being equal to 9.75 m^2. Obtained this result, the design proceeds with the calculation of the airflow required by the conventional aerothermal system and the electric power needed by the innovative configuration. In the first case, assuming the air temperature of the pneumatic system equal to 250°C, the obtained airflow is 0.41 kg/s. In the second case, the total electric power exceeds 25.9 kW, as a total of 52 cyclically protected zones are determined by the optimization algorithm. In both conventional and innovative architectures, the ice protection of the small users requires an electrical power of about 10 kW.

Then, the design process proceeds with the preliminary sizing of the air conditioning system. The four kinds of thermal loads evaluated in the conditions of maximum heating and maximum cooling requests are reported in Table 22.

Table 22: Thermal loads of the regional jet.

Thermal load	**Max heating request**	**Max cooling request**
Heat flow through fuselage	81.6 kW	-10.5 kW
Solar heating	0 kW	- 2.07 kW
People physiological heating	- 0.26 kW	- 10.68 kW
Cabin equipment heating	- 8.38 kW	- 8.38 kW
Total thermal load	*72.96 kW*	*-31.63 kW*

According to eq. 7, the airflow entering the cabin to maintain an environment temperature within the range 18°C – 25°C might reach 3.28 kg/s. Two subfreezing ACMs are selected to be installed within the air conditioning system. These two CAUs receive a maximum airflow of 1.64 kg/s, considering 50% of percentage of recirculation.

Once all the utility systems have been sized, the power generation and distribution systems are designed. The “CONV” architecture is characterized by a standard 3000 psi HPGDS, with hydraulic pumps driven by the two jet engines through the AGBs. The other architecture with hydraulic system is “MEA 2”. However, in this case the hydraulic oil pressure is set to 5000 psi, while the hydraulic pumps are moved by dedicated electric motors. The total hydraulic – or electric, in case of electric driven pumps – power is evaluated per each mission segment. More details are provided in the following subsections. Concerning the PPGDS, the first two architectures are traditional, as the airflow is bled from the engines at a maximum pressure of 7 bar and a maximum temperature of 250°C. The airflow is then delivered to the ducts of the wing de-ice system and to the air conditioning packs. Again, the airflow budget is evaluated according to the quantity of hot and high pressure air required by the utility subsystems. These airflow budgets are collected and described in the following subsections. The other two architectures are “bleedless”. Thus, the airflow necessary for the ECS is gathered from the external environment and pressurized by two dedicated compressors driven by electric motors. The electric power needed for the compression is evaluated by means of eq. 9, where the airflow $\dot{m}_{PN}$ passing through the compressor and the external air temperature T_{ext} depend on the mission phase. The efficiencies

η_{compr} and η_{elect} are set both to 0.8. Finally, the electric system is sized for each on-board systems architecture. The conventional configuration generates 115 V AC (400 Hz) electric power by means of three IDGs, two moved by the engines and one by the APU. Part of this power is then converted to 28 V DC to supply all the users alimented by low voltage direct current. All the other architectures are characterized by high voltage electric power systems. In particular, the total electric power is generated by two 235 V AC permanent magnet alternators per engine and APU. The electric power is then partly converted in 150 V AC, 270 V DC and 28 V DC. As the HPGDS and the PPGDS, the global electric system power budgets are later on presented.

4.4.3 On-board systems impact on the OAD

In subsection 4.4.2 the preliminary design of the regional jet on-board systems has been described. Now, the mass and power off-takes results concerning the four previously introduced subsystem architectures are reported and discussed. In particular, in the current subsection the attention is not posed only on subsystems level, but also on the entire aircraft level. The effects of the four kinds of subsystem architecture on the OAD are highlighted, mainly showing the impacts on variation of aircraft empty weight and fuel consumption.

The 1st Generation MDO framework proposed in subsection 3.5.1 is first calibrated for the design of the conventional regional jet, characterized by the conventional on-board systems architecture. For the calibration of the design framework, the aircraft results collected in [194] are employed. These results slightly differ from those reported in Table 19, as they derive from a design process in which the results have been refined by further iterations. The comparison of some design outputs are reported in Table 23.

Table 23: Calibration of the 1st Generation MDO framework.

Parameters	1st Gen MDO framework	3rd Gen MDO framework	Relative difference [%]
MTOM [kg]	42269	43332	2.5
OEM [kg]	22546	23965	5.9
Fuel mass @ max payload [kg]	8223	7867	-4.5
Wing area [m^2]	71.4	70.4	-1.4
Wing span [m]	26	25.86	-0.5
Max engine net thrust [kN]	85.2	90	5.3

It can be noted that the relative difference is in the range [-4.5%; 5.9%]. These values are considered acceptable to validate the calibration of the 1st Generation MDO framework.

Once the 1st Generation MDO framework has been calibrated, four design cases are conducted, one per each on-board systems architecture. The Table 24 collects the masses of the aircraft subsystems.

Table 24: Aircraft on-board systems mass budget.

	CONV	**MEA 1**	**MEA 2**	**AEA**
Avionic system mass [kg]	617	617	617	617
Flight Control System mass [kg]	572	743	572	743
Landing gear system mass [kg]	1281	1306	1279	1311
ECS&IPS (+ pneumatics) mass [kg]	648	647	615	615
Fuel system mass [kg]	383	382	382	382
APU system mass [kg]	169	167	169	168
Furnishing mass [kg]	3083	3081	3083	3082
Hydraulic System mass [kg]	497	0	356	0
Electric System mass [kg]	954	1009	1174	1208
Total systems mass [kg]	*8204*	*7952*	*8247*	*8126*

Part of the results collected in Table 24 are also shown in the form of bar chart, as represented in Figure 53. In the diagram, only subsystems impacted by the different levels of technology are shown. The mass of the FCS increases for the architectures characterized by innovative actuators, in particular EHAs and EMAs. The same happens for the landing gear. However, in this case the actuator mass represents a little percentage of the entire subsystem mass. Therefore, the four system weights are similar. The third item represents the total mass of air conditioning and de-icing systems. In this entry, the mass of the pneumatic system – if present – is included. The results show a slight decrease of this mass moving from a traditional to a bleedless configuration. This mass decrement encompasses the introduction of new components (i.e. the dedicated compressors), the electrification of the WIPS and the removal of the pneumatic system. More remarkable differences regard the HPGDS. The “MEA1” and “AEA” architectures are characterized by the elimination of the hydraulic system, as represented by the null value of the mass. The third architecture differs from the conventional one due to the increase of the hydraulic oil pressure, therefore entailing a reduction of the system mass. Finally, the electric system mass derives from the total electric power (which increases with the four architectures), the electric voltage and the typology of electric machines. Thus, the mass of the EPGDS shows the trend depicted in Figure 53. As previously aforementioned, the results of Table 24 and Figure 53 are obtained by means of the 1st Generation MDO framework. The same analysis is done through the AGILE's innovative design environment, obtaining similar results. In particular, the main differences regard a limited number of subsystems, and they are due to the discrepancies collected in Table 23. However, the trend shown in Figure 53 is analogous to the trend resulting from the design performed with the AGILE's 3rd Generation MDO framework.

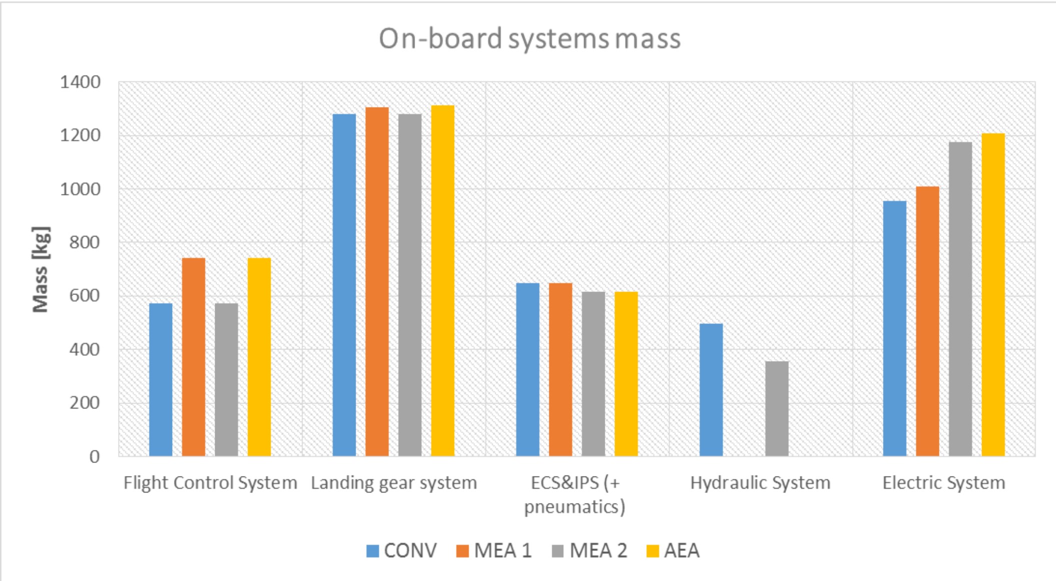


Figure 53: Subsystems mass comparison for the four on-board system architectures.

The secondary power level is then estimated per each on-board system architecture. It should be noted that two levels of secondary power can be evaluated. One level is the installed power [21], which is needed to size components as electric generators and hydraulic pumps. However, in order to estimate the amount of fuel required by subsystems, only the secondary power actually consumed during the mission profile should be accounted. For sake of simplicity, in this dissertation only the installed power is calculated, which will be also used afterwards for the estimation of the impact on the engines efficiency during the mission profile and hence on the fuel consumption. The Table 25, Table 26 and Table 27 collect the engines shaft power and bleed air off-takes during all the phases of the mission. In particular, a typical flight mission profile is defined, in which additional sub-phases are included. These sub-phases are enclosed within the typical flight phases, for instance take-off, climb and cruise. During the sub-phases, a surplus of secondary power might be required for a limited period of time. Therefore, in the current mission profile are included sub-phases during which the landing gear and the flaps are actuated.

In more details, the following tables report the shaft power required by the electric and the hydraulic systems and the bleed air tapped from the engines compressors. Thus, mission phases during which the engines are turned off are not considered.

Table 25: Regional jet electric shaft power off-takes.

Mission segment	**Electric shaft power off-takes [kW]**			
	CONV	MEA1	MEA2	AEA
Taxi out	37.5	40.2	94.3	93.9
--> *Flaps extension*	*37.5*	*43.6*	*98.1*	*97.3*
Take-off	35.5	55.5	112.0	109.3
--> *Landing gear retraction*	*35.5*	*57.8*	*114.7*	*111.6*
--> *Flaps retraction*	*35.5*	*58.8*	*115.9*	*112.7*
Climb	50.1	70.1	189.0	186.3
Cruise	49.9	70.0	200.2	197.5
Descent	50.1	70.1	189.0	186.3
Approach	49.9	80.5	151.2	147.0
--> *Flaps extension*	*49.9*	*73.3*	*147.3*	*144.1*
--> *Landing gear extension*	*49.9*	*82.9*	*153.9*	*149.4*
Landing	34.9	70.9	129.5	124.6
Taxi in	37.5	40.2	94.3	93.9
--> *Flaps retraction*	*37.5*	*43.6*	*98.1*	*97.3*

Table 26: Regional jet hydraulic shaft power off-takes.

Mission segment	**Hydraulic shaft power off-takes [kW]**			
	CONV	MEA1	MEA2	AEA
Taxi out	3.2	0	0	0
--> *Flaps extension*	*7.0*	*0*	*0*	*0*
Take-off	22.8	0	0	0
--> *Landing gear retraction*	*25.4*	*0*	*0*	*0*
--> *Flaps retraction*	*26.6*	*0*	*0*	*0*
Climb	22.8	0	0	0
Cruise	22.8	0	0	0
Descent	22.8	0	0	0
Approach	34.8	0	0	0
--> *Flaps extension*	*37.5*	*0*	*0*	*0*
--> *Landing gear extension*	*26.6*	*0*	*0*	*0*
Landing	39.1	0	0	0
Taxi in	3.2	0	0	0
--> *Flaps retraction*	*7.0*	*0*	*0*	*0*

Table 27: Regional jet electric bleed air off-takes.

Mission segment	Bleed air off-takes [kg/s]			
	CONV	MEA1	MEA2	AEA
Taxi out	1.83	1.83	0	0
--> *Flaps extension*	*1.83*	*1.83*	*0*	*0*
Take-off	0	0	0	0
--> *Landing gear retraction*	*0*	*0*	*0*	*0*
--> *Flaps retraction*	*0*	*0*	*0*	*0*
Climb	1.21	1.21	0	0
Cruise	0.80	0.80	0	0
Descent	1.21	1.21	0	0
Approach	1.56	1.56	0	0
--> *Flaps extension*	*1.56*	*1.56*	*0*	*0*
--> *Landing gear extension*	*1.56*	*1.56*	*0*	*0*
Landing	0	0	0	0
Taxi in	1.83	1.83	0	0
--> *Flaps retraction*	*1.83*	*1.83*	*0*	*0*

The same results are also provided in form of histograms, as represented in Figure 54, Figure 55 and Figure 56. It is immediately possible to note that not all the subsystem architectures generate all the three kinds of power. The electric power off-takes are indeed present on all the configurations, while the hydraulic power is generated only by the conventional solution. The architecture “MEA1” is in fact characterized by hydraulic pumps driven by electric motors, while the other two innovative configurations haven’t the hydraulic system. Concerning the pneumatic power, the bleed air off-takes are null in the bleedless architectures, i.e. “MEA2” and “AEA”.

The Figure 54 shows the electric shaft power off-takes required by the four on-board system architectures. The electric power request increases passing from the conventional architecture to the most innovative ones, due to the electrification of the FCS (“MEA1” and “AEA”) and the ECS and WIPS (“MEA2” and “AEA”). Moreover, it can be seen that from the climb to the descent phases it is required more electric power, as several electric users are supposed to be turned on, as galley ovens. Furthermore, in the case of all the architectures except the conventional one, it is worth noting the increment of power during the sub-phases, in which additional electric power is required to actuate the secondary mobile surfaces and the landing gear.

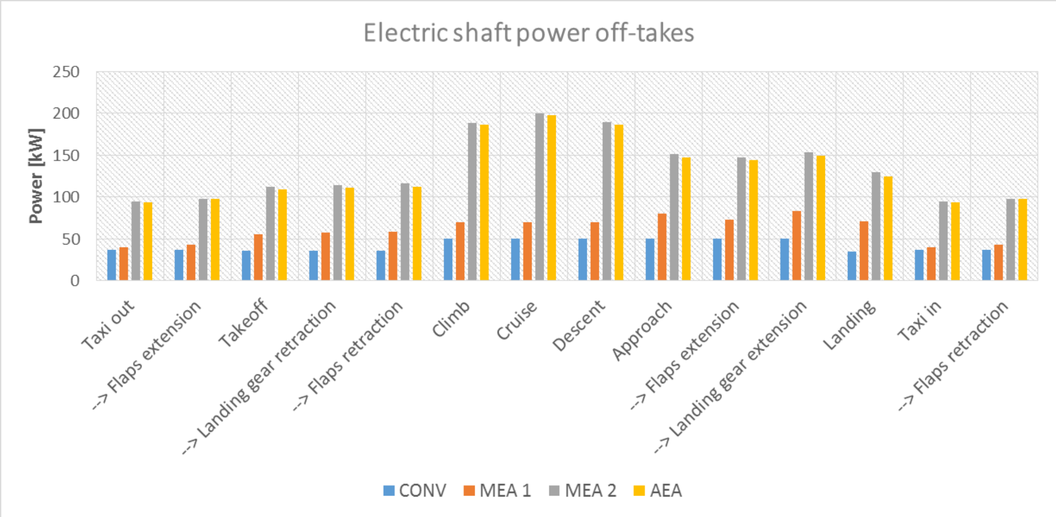


Figure 54: Regional jet electric shaft power off-takes.

The increment of secondary power during the sub-phases is depicted also in Figure 55, which represents the hydraulic shaft power off-takes of the conventional architectures. From this graph is evident the high request of hydraulic power during the approach and the landing phases. In the first case, the speed airbrakes are employed to reduce the airplane speed. In the second case, ground spoilers are utilized to break the lift force and increase the aerodynamic drag, while landing gear brakes are actuated to decelerate the airplane on the runway.

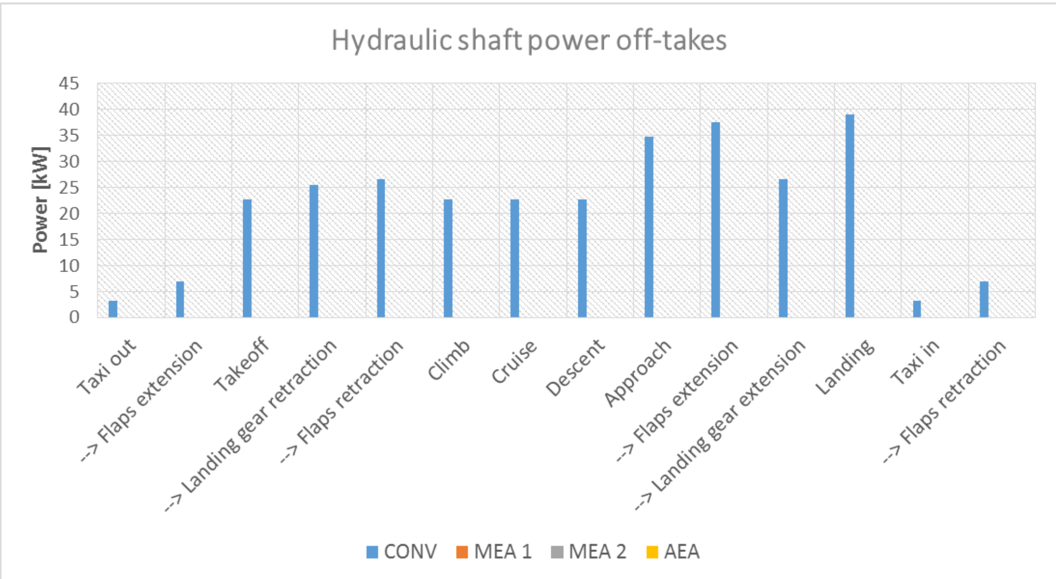


Figure 55: Regional jet hydraulic shaft power off-takes.

The last graph (Figure 56) depicts the bleed air supplied to the aerothermal wing de-ice system and to the air conditioning system. In take-off and landing the air conditioning and de-ice systems are turned off, with decrement of the bleed air off-take. The reduction of required airflow in climb and descent results from the different thermal loads acting in these phases. The bleed air off-take during cruise is instead decreased as the wing de-ice system is inactive.

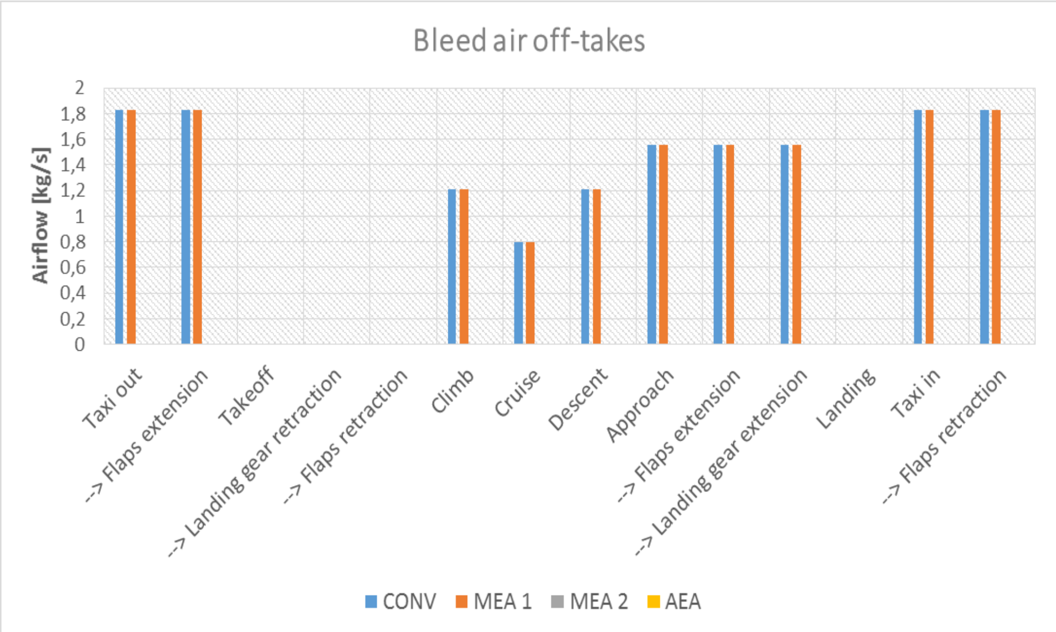


Figure 56: Regional jet bleed air off-takes.

As the case of the system mass estimation, the determination of the power off-takes has derived by means of the 1st Generation MDO framework set up in the context of the doctoral research. Employing the AGILE's 3rd Generation MDO framework, analogous results are calculated.

In conclusion of the current subsection, a comparison among the global aircraft masses – namely the OEM, the fuel mass and the MTOM – estimated through the two kinds of MDO frameworks is presented. These results are provided in Table 28 and Figure 57.

Table 28: Regional jet global masses estimated through the 1st and the 3rd Generation MDO frameworks.

	1st Generation				3rd Generation			
	CONV	**MEA1**	**MEA2**	**AEA**	**CONV**	**MEA1**	**MEA2**	**AEA**
OEM [kg]	22546	22110	22563	22354	23965	23762	23929	23807
Fuel mass [kg]	8223	8139	8134	8093	7867	7796	7522	7497
MTOM [kg]	42269	41749	42197	41947	43332	43058	42951	42804

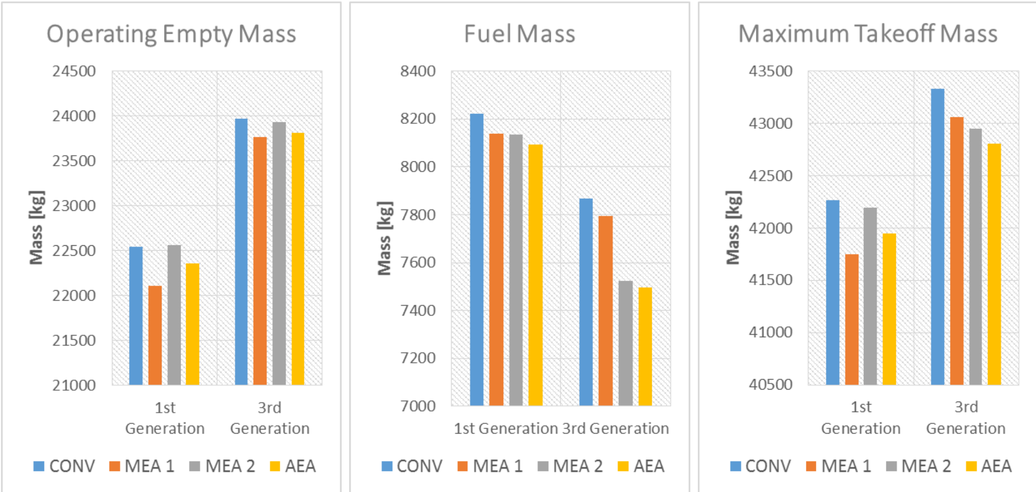


Figure 57: Regional jet global masses estimated through the 1st and the 3rd Generation MDO frameworks: (a) OEM; (b) Fuel mass and (c) MTOM.

It can be seen that the results obtained by means of the two frameworks are not completely coincident, as the fidelity level of the involved tools is different. In other words, the 3rd Generation MDO framework employs codes developed by experts with a deep competence in the own discipline. Therefore, the results obtained with this kind of collaborative and distributed environment are more reliable. However, the masses derived with the 1st Generation framework present similar trends respect to those obtained by the other MDO framework, except for few cases (for instance, the MTOM of “MEA1”). Thus, this fact confirms what stated in Section 2.4, i.e. the 1st Generation framework might be utilized for a fast overall aircraft conceptual design.

4.4.4 DOE of on-board system architectures

The same reference aircraft described in subsection 4.4.1 is utilized for an additional study: the analysis and design of more than 120 different on-board system architectures. In other words, the design space is deeply investigated considering different combinations of the design variables – e.g. electric system voltage, bleed/bleedless architecture, hydraulic oil pressure – of the four subsystem configurations previously treated. Knowing the effects of these architectures on other design disciplines might be very important for the selection of the optimal subsystem configuration since the conceptual design phase.

Other studies present in literature are devoted to the derivation and design of different subsystem architectures. One of these studies is carried on by Judt *et al* [200], determining several combinations of a regeneration energy system for a High Altitude Long Endurance (HALE) UAV. In [18] instead the focus is centered on the development of More and All Electric system architectures, defining over 13 million possible configurations. In this dissertation however the maximum number of subsystem architectures is set to 124, varying the parameters reported in Table 29.

Table 29: Alternative design options for the different architectures.

Subsystem	Options
Flight Control System	Hydraulic, electric
Landing gear – retraction system	Hydraulic, electric
Landing gear – steering system	Hydraulic, electric
Landing gear – braking system	Hydraulic, electric
Hydraulic system pressure	3000 psi, 5000 psi
Primary electric voltage	115 V AC (400 Hz), 235 V AC wf
Pneumatic system configuration	Conventional, bleedless

It is worth noting that from a mathematical perspective the total number of possible solutions is 128, but not all the configurations are reasonable from an engineering point of view. In the solutions characterized by the removal of the hydraulic system, the design variable concerning the hydraulic system pressure becomes worthless. A subset of all the feasible architectures is listed in Table 30. It can be noted that the four subsystem architectures treated in subsection 4.4.1 are included within the 124 combinations (CONV: #117; MEA1: #3; MEA2: #124; AEA: #4).

Table 30: Subset of the 124 on-board system architectures.

#	FCS [hydr/elec]	Landing gear [hydr/elec]			Hydraulic press [psi]	Electric volt [115V/235V]	Bleedless [yes/no]
		Retract	**Steering**	**Braking**			
3	elec	elec	elec	elec	-	235 V	no
4	elec	elec	elec	elec	-	235 V	yes
6	elec	elec	elec	hydr	3000	115 V	yes
14	elec	elec	hydr	elec	3000	115 V	yes
54	elec	hydr	hydr	hydr	3000	115 V	yes
61	hydr	elec	elec	elec	3000	115 V	no
64	hydr	elec	elec	elec	3000	235 V	yes
73	hydr	elec	elec	hydr	5000	115 V	no
75	hydr	elec	elec	hydr	5000	235 V	no
79	hydr	elec	hydr	elec	3000	235 V	no
89	hydr	elec	hydr	hydr	5000	115 V	no
91	hydr	elec	hydr	hydr	5000	235 V	no
97	hydr	hydr	elec	elec	5000	115 V	no
99	hydr	hydr	elec	elec	5000	235 V	no
107	hydr	hydr	elec	hydr	5000	235 V	no
116	hydr	hydr	hydr	elec	5000	235 V	yes
117	hydr	hydr	hydr	hydr	3000	115 V	no
121	hydr	hydr	hydr	hydr	5000	115 V	no
123	hydr	hydr	hydr	hydr	5000	235 V	no
124	hydr	hydr	hydr	hydr	5000	235 V	yes

All the 124 architectures are sized through the on-board systems design tool ASTRID, obtaining different system masses and power off-takes. As depicted in the workflow schema represented in Figure 58, the power off-takes are given to an engine RSM, which estimates the fuel consumed by the on-board systems. The engine RSM is a surrogate model of an engine deck based on the commercial tool GasTurb v12 ([87], [88]) for engine preliminary design and performance simulation. As this software requires high complex and high time demanding operations, a RSM is developed to speed up the optimization process without sacrificing the accuracy of the results.

From the flowchart of the workflow it can be noted that the “snowball” effect is neglected, otherwise several additional disciplinary design modules should be implemented. However, in the conclusive part of the current subsection simple equations for the preliminary quantification of the “snowball” effect will be presented.

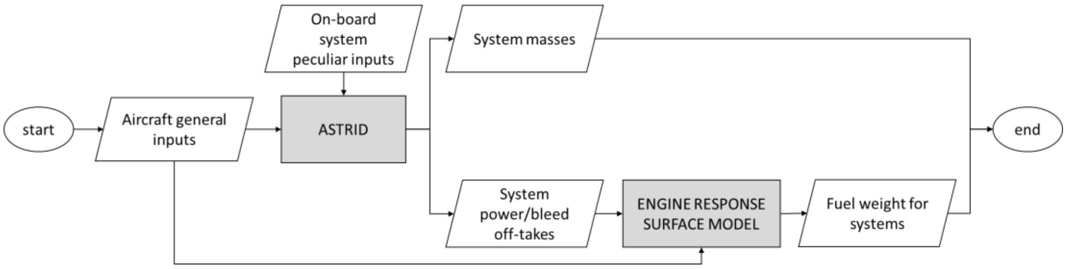


Figure 58: Flowchart of the workflow for the design of the 124 subsystem architectures (adapted from [201]).

The results concerning some architectures among those listed in Table 30 are reported in Table 31 and Table 32, in particular showing per each configuration the resulting systems mass, the fuel consumed by subsystems, and the sum of these two values. It is worth noting that these results – in particular concerning the on-board systems mass – are slightly different from those collected in Table 24, Table 25, Table 26 and Table 27. Differently from the results reported in subsection 4.4.3, those of Table 31 are derived from an un-converged MDA problem.

Table 31: Subset of DOE results: systems mass increasing.

#	Systems mass [kg]	Shaft power off-takes [kW]	Bleed air off-takes [kg/s]	Systems fuel [kg]	Total mass [kg]
123	8071	72.7	0.7968	120.4	8191
107	8074	72.7	0.7968	120.4	8195
3	8079	70	0.7968	119.6	8199
…	…	…	…	…	…
75	8115	72.7	0.7968	120.4	8236
4	8119	197.5	0	52.3	8172
73	8128	72.7	0.7968	120.4	8248
…	…	…	…	…	…
116	8176	200.2	0	53.2	8229
124	8176	200.2	0	53.2	8229
89	8185	72.7	0.7968	120.4	8306
…	…	…	…	…	…
79	8253	72.7	0.7968	120.4	8374
117	8259	72.7	0.7968	120.4	8379
61	8273	72.7	0.7968	120.4	8394
…	…	…	…	…	…
54	8679	197.5	0	52.3	8731
6	8719	197.5	0	52.3	8772
14	8719	197.5	0	52.3	8772

Table 32: Subset of DOE results: total mass increasing.

#	Systems mass [kg]	Shaft power off-takes [kW]	Bleed air off-takes [kg/s]	Systems fuel [kg]	Total mass [kg]
4	8119	197.5	0	52.3	8172
123	8071	72.7	0.7968	120.4	8191
107	8074	72.7	0.7968	120.4	8195
3	8079	70	0.7968	119.6	8199
121	8081	72.7	0.7968	120.4	8202
99	8093	72.7	0.7968	120.4	8213
…	…	…	…	…	…
116	8176	200.2	0	53.2	8229
124	8176	200.2	0	53.2	8229
97	8113	72.7	0.7968	120.4	8233
91	8115	72.7	0.7968	120.4	8236
…	…	…	…	…	…
79	8253	72.7	0.7968	120.4	8374
117	8259	72.7	0.7968	120.4	8379
64	8336	200.2	0	53.2	8389
…	…	…	…	…	…
54	8679	197.5	0	52.3	8731
6	8719	197.5	0	52.3	8772
14	8719	197.5	0	52.3	8772

From the results reported in Table 31 and Table 32, several considerations can be done. Considering only the resulting systems mass, the heavier solutions are characterized by the bleedless configuration, by the lower hydraulic oil pressure and by the lower generated electric voltage. The electrification of the WIPS and ECS entails reduction in their masses, but also an enlargement of the electric system. On the other side, higher hydraulic oil pressures and higher electric voltages involve weight reductions. It is worth noting that the lightest on-board system architecture is MEA1 (#3), while the heaviest is represented by CONV (#117). This trend is in line with what shown in Table 24 and Figure 53.

Concerning the total mass, again the higher electric voltage and the higher hydraulic system pressure entail lightest architectures. However, the bleedless configuration brings to a deep reduction of the fuel consumed by subsystems. Therefore, some solutions with electric ECS and WIPS are included within the lightest architectures. Among these solutions, the AEA (#4) results the lightest one, while the CONV (#117) is the heaviest among the four architectures.

The results of the DOE of on-board system architectures entail the assessment of the system masses of several configurations, and the impacts of these in terms of fuel consumption. However, the effects on other aircraft global variables – as MTOM and OEM – are not evaluated. Moreover, the "snowball" effect can't be captured. But, during the aircraft conceptual design phase it could be extremely useful to preliminarily assess the impacts of the on-board system architectures on the airplane global variables. It is in the conceptual design phase that the subsystem architecture shall be selected. Therefore, it is important to correctly evaluate the effects of the configurations on the entire aircraft, in order to select the "best" one. In addition, during the conceptual design phase low fidelity codes are available and the time and effort allocated are limited. Thus, it is required to quickly and approximately assess the impacts of all the possible on-board system architectures on the entire aircraft. It is here proposed a RSM of the entire OAD process implemented in the 1st Generation MDO framework described in subsection 3.5.1. This RSM has been created from a DOE of 275 experiments. Therefore, a full-factorial design space sampling method is chosen, considering all the possible combinations of different values of system masses, shaft power off-takes and bleed air off-takes. The DOE inputs list is created on the basis of the ranges of values collected in Table 33. The reference values and their variation ranges are partly derived from the results of the DOE of subsystem architectures and partly taken from previous studies carried on by the research team ([201], [202]). These values are referred to the conventional on-board system architecture (#117).

Table 33: Main subsystems results and their considered variation (adapted from [203]).

On-board systems results	Reference value	Variation range
Systems mass (M_{sys})	8379 kg	-4% ÷ 4% (~8044 kg ÷ 8714 kg)
Systems power off-takes ($P_{offtakes}$)	72.7 kW	0% ÷ 200% (72.7 kW ÷ 218.1 kW)
Systems bleed air off-takes (P_{bleed})	0.7968 kg/s (cruise)	0% ÷ -100% (0.7968 kg/s ÷ 0 kg/s)

For each one of the 275 design experiments, a converged solution is obtained, i.e. an entire aircraft is designed. A first-order polynomials model is therefore constructed. This response model is an approximate representation of the OAD process.

Thus, a surface response characterized by the equations eq. 39, eq. 40 and eq. 41 is derived:

$$MTOM^{*} = 25120 + 2.0524 \cdot M_{sys} + 0.00082 \cdot P_{offtakes} + 242.6 \cdot P_{bleed} \quad \text{eq. 39}$$

$$Fuel^{*} = 5322 + 0.3291 \cdot M_{sys} + 0.00053 \cdot P_{offtakes} + 156.9 \cdot P_{bleed} \quad \text{eq. 40}$$

$$OEM^{*} = 8298.4 + 1.7233 \cdot M_{sys} + 0.00029 \cdot P_{offtakes} + 85.67 \cdot P_{bleed} \quad \text{eq. 41}$$

Applying these equations to the results of the on-board systems design discipline, considering as reference case an aircraft similar to the AGILE project regional jet, a preliminary evaluation of the main OAD parameters can be obtained. Some of these results are collected in Table 34. The results calculated by means of the OAD process (subsection 4.4.3) are also reported and compared. The comparison among these results validates the RSM, as the percentage difference of the resulting main parameters is very low (less than ±0.5%).

Table 34: Validation of the OAD response model.

	CONV	**MEA1**	**MEA2**	**AEA**
MTOM (RSM) [kg]	42211	41692	42210	41959
MTOM (framework) [kg]	42269	41749	42197	41947
Difference	*0.14%*	*0.14%*	*-0.03%*	*-0.03%*
Fuel mass (RSM) [kg]	8185	8101	8142	8101
Fuel mass (framework) [kg]	8223	8139	8134	8093
Difference	*0.46%*	*0.47%*	*-0.1%*	*-0.09%*
OEM (RSM) [kg]	22526	22091	22568	22359
OEM (framework) [kg]	22546	22110	22563	22354
Difference	*0.10%*	*0.09%*	*-0.02%*	*-0.02%*

Once the OAD response model is validated, it can be employed for the preliminary evaluation of the main results of the OAD process. In Table 35 are reported the estimations of MTOM, fuel mass and OEM of different designs characterized by some of the 124 previously defined on-board system architectures.

Before concluding the current subsection, it is worth noting that the built RSM becomes useful in case a very high number of subsystem architectures shall be designed and compared. In the current dissertation only 124 on-board system configurations are identified, as the main purpose is to demonstrate the feasibility of the proposed method. However, as evident in [18], a very higher number of possible architectures can be defined, making indispensable the employment of a RSM to preliminarily assess the main parameters of all the designs.

Table 35: Application of the OAD response model.

#	Estimated MTOM [kg]	Estimated Fuel mass [kg]	Estimated OEM [kg]
3	41952	8143	22309
4	41945	8098	22348
6	43177	8296	23382
14	43177	8296	23382
54	43093	8282	23312
61	42353	8208	22645
64	42392	8171	22722
73	42054	8160	22395
75	42029	8156	22373
79	42311	8202	22610
89	42172	8179	22493
91	42029	8156	22373
97	42023	8155	22368
99	41982	8149	22334
107	41944	8143	22302
116	42063	8118	22445
117	42323	8203	22620
121	41959	8145	22314
123	41937	8142	22296
124	42063	8118	22445

4.5 Design-To-Cost applied to hybrid powered airplanes

The last case study presented in the current dissertation concerns an application of the DTC approach for the development of a hybrid propulsion system to be installed aboard the same reference UAV of Section 4.2.

As explained in Section 2.3, according to the DTC approach, the parameter "cost" is considered as a high level requirement. Therefore, the product shall be designed in compliance with the target cost defined at the beginning of the development process. In other words, all the design choices are made to meet all the requirements, included the cost. It is worth noting that the term "cost" here refers to the development and production cost of a single product. However, the design process shall consider also the overall LCC of the airplane. At the end of the development process, the final product might be compliant with the predetermined development and production cost. But, excessive operating costs of the aircraft –

e.g. fuel and maintenance costs – will make the product uncompetitive with other airplanes of the same class. In the following case study, only the development and production cost will be considered as a design requirement.

In the proposed case study, several propulsion system architectures are defined, designed and compared. All the architectures are hybrid retrofits of a reference UAV, but they differ in terms of system capabilities and functionalities. The six-step methodology proposed in Section 3.7 is adopted in the present application study.

4.5.1 Cost estimation of the traditional propulsion system

The development process described in Chapter 3 is adopted for the preliminary design of a traditional propulsion system installed aboard the reference unmanned airplane introduced in Section 4.2.

A subset of the requirements at air-vehicle level is reported in Table 36. For sake of clarity, only requirements linked to the propulsion system are collected.

Table 36: Subset of high level requirements of the air-vehicle equipped with a traditional propulsion system.

Air-vehicle requirement	Type
The air-vehicle shall generate propulsive power only through an endothermic source	Functional
The air-vehicle shall generate and distribute electric power	Functional
The air-vehicle shall be started autonomously	Functional
The air-vehicle shall be started by means of a ground cart	Functional
The air-vehicle shall be fueled with JET A-1	Architectural
The air-vehicle shall be fueled with Diesel Fuel	Architectural
The air-vehicle shall have a ceiling altitude of at least 7500 m	Performance
The air-vehicle shall fly at a cruise altitude of at least 7000 m	Performance
The air-vehicle shall have an operational speed of at least 250 km/h (TAS) at 7000 m	Performance
The air-vehicle shall take-off from a runway with length of at least 1500 m (ISA+20°C; runway elevation: 1000 m)	Performance

A functional model is developed from the functional requirements. The activity diagram of the UC “Provide propulsive power” is shown in Figure 59 as an example.

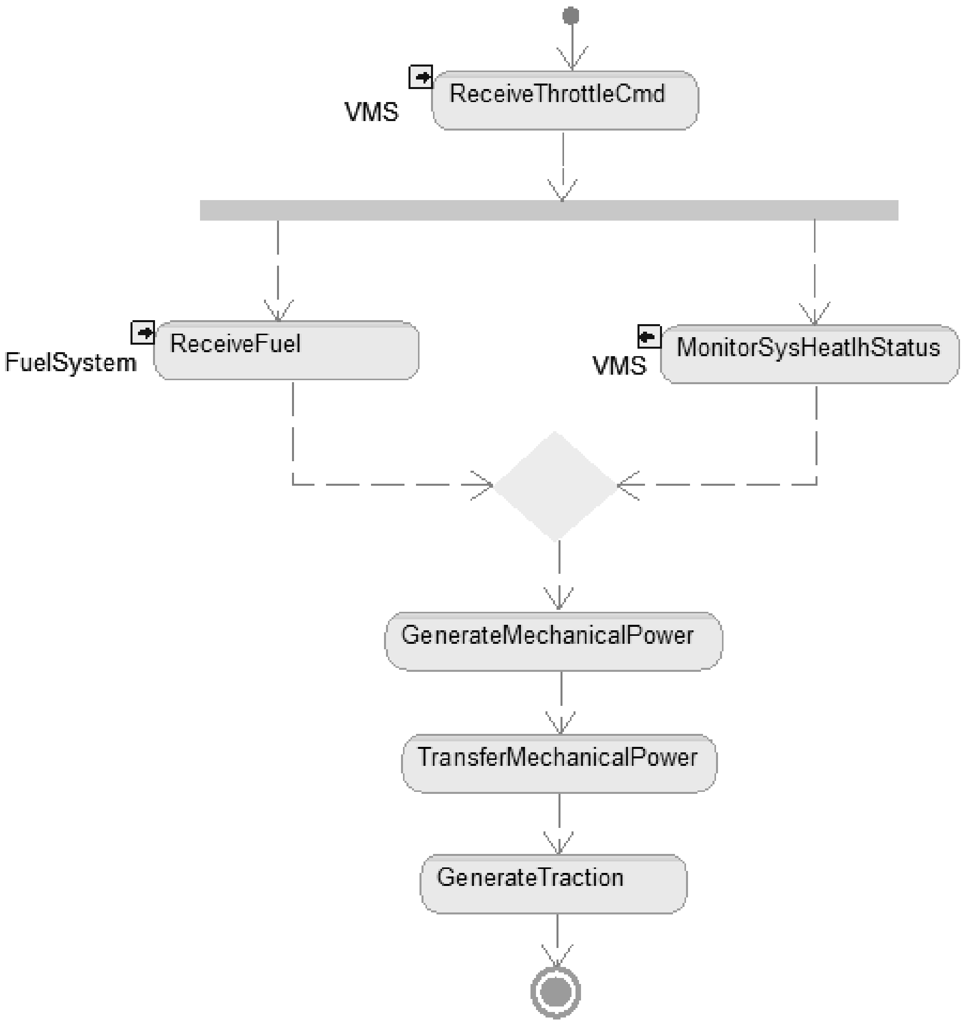


Figure 59: Black-Box Activity Diagram of the UC "Provide propulsive power" relative to the design of the traditional propulsion system.

The reader could note the similarity of this flow-chart with the diagram depicted in Figure 35, relative to the design of the hybrid propulsion system. Indeed, in this case study, only the operating mode "Traditional" is considered. Therefore, the branch of the Activity Diagram relative to the operating modes "Powerboost", "ElectricFlight" and "GreenTaxi" is removed.

Then, the components identified through the functional model are sized. A 1st Generation MDO framework is adopted to perform the preliminary design of the entire UAV. Therefore, the design procedure is focused on the propulsive system level, obtaining the results reported in Table 37.

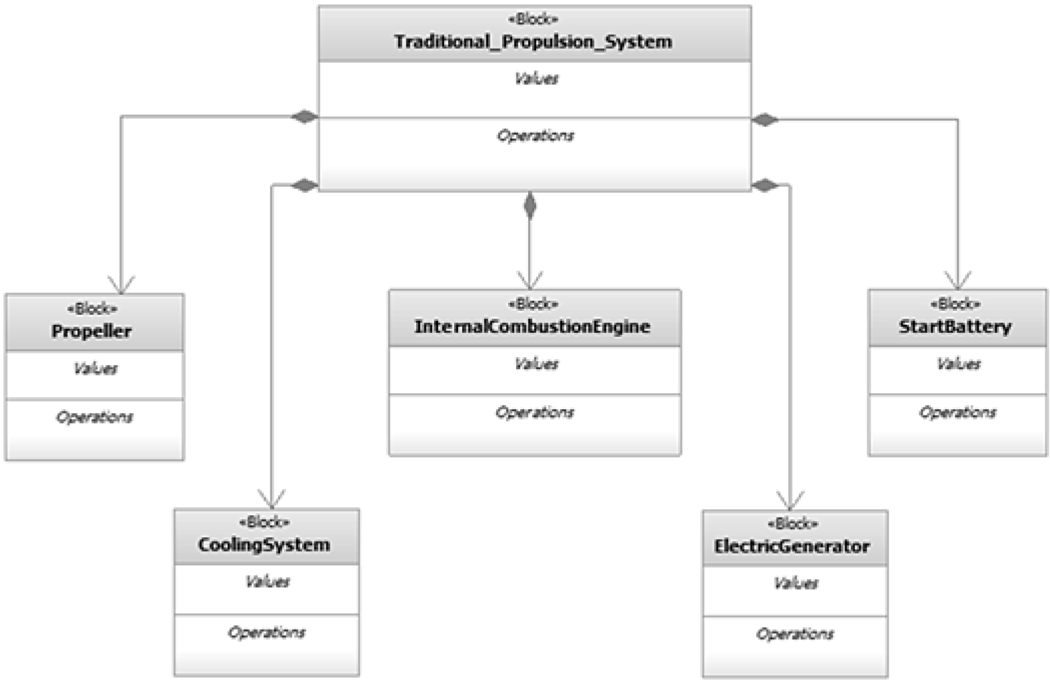


Figure 60: BDD of the traditional propulsion system.

Table 37: Subset of the preliminary design results of components of the traditional propulsion system.

Component	**Features**
Propeller	Max traction: 5000 N; Mass: 25 kg
ICE	Max power: 370 kW; Mass: 330 kg
Battery	Energy: 300 Wh; Mass: 5 kg
Cooling system	Mass: 75 kg
Electric generator	Max power: 10 kW; Mass: 30 kg

The results collected in Table 37 are employed for the estimation of the development and production cost of each component. The cost of the equipment is evaluated by means of a commercial software – PRICE TruePlanning®[5] [204] – in which CERs are coded for the parametric cost estimation. The obtained costs are

[5] PRICE TruePlanning is a commercial cost estimation software supported by a large database collecting data gathered from decades of most complex aerospace, defence, and security applications.

reported in Table 38, in which modification and integration costs are included and allocated inside the cost of each component. Thus, the total cost estimate of an entire single traditional propulsion system results of k€ 300.

Table 38: Cost of the components of the traditional propulsion system.

Component	Cost [2017 k€]
Propeller	25
ICE	185
Battery	5
Cooling system	5
Electric generator	80
TOTAL	**300**

4.5.2 Definition of the target cost

The following step refers to the definition of the target cost. As explained in Section 2.3, the target cost derives from a market survey or it is assessed on the basis of how much the customer is willing to pay. However, in the current case study the target cost is assumed one third higher than the cost of the traditional propulsion system. Therefore, the defined cost of the hybrid-electric propulsion system shall not exceed k€ 400. This target cost is hence considered as a system requirement.

4.5.3 Identification of the alternative system architectures

Five alternative system architectures are identified. The architectures differ in terms of capabilities and functionalities. A MBSE approach with reference to the functional model presented in Section 4.2 is adopted to derive the five solutions.

The first hybrid-electric architecture is characterized by the five capabilities described in Section 4.2. The system shall generate propulsive and secondary power by means of a traditional endothermic source. However, the taxi phase shall be performed electrically, while during the take-off an electric source shall assist the endothermic source for the generation of the propulsive power. During the descent, energy shall be recovered for the generation of electric power, which shall be stored and utilized later to move the propeller, hence smoothing the descent trajectory. Furthermore, the system shall be automatically managed by a system controller. This architecture is named "Hyb_Sys". The second architecture derives from the first one. In this alternative solution, the operating mode "RATmode" is removed. Therefore, the system might not employ the propeller to recover energy. However, if needed, the system shall provide propulsive power during the flight through only the electric source. This second solution is named "Hyb_No_RAT". Removing the capability of performing the electric taxi, a third architecture is identified. All the other operating modes are considered. This architecture is named "Hyb_No_Taxi". The fourth system architecture – named "Hyb_No_Aut" – has all the capabilities of the first solution. However, this architecture is characterized by a lower level of autonomy, as many functions are performed by the ground operator instead of the system controller. The last architecture is a mixture of the third and fourth solutions. The "GreenTaxi" operating mode is removed, together with all the functions that make the system autonomous. Therefore, the ground operator shall perform several actions in place of the system controller. The last architecture is named "Hyb_No_Taxi_No_Aut".

Once the several system architectures are identified, the case study proceeds with the scoring of the solutions. In other words, the various solutions are prioritized according to the customer needs and expectations. Therefore, in Table 39 are reported all the scores per each operating mode for all the system architectures. The traditional propulsion system architecture (named "Trad_Sys") is evaluated, too.

Table 39: Importance of the capabilities of the five alternative architectures.

	System architecture					
Operating mode	Hyb_Sys	Hyb_No _RAT	Hyb_No _Taxi	Hyb_No _Aut	Hyb_No_Taxi_ No_Aut	Trad_ Sys
Powerboost	20	20	20	15	15	0
Traditional	20	20	20	15	15	20
GreenTaxi	20	20	0	15	0	0
RATmode	20	0	20	15	15	0
ElectricFlight	20	20	20	15	15	0
TOTAL	**100**	**80**	**80**	**75**	**60**	**20**

The assignment of the scores to each alternative solution is based on a customer interviewing process. Several techniques are present in literature with the aim of evaluate through scores qualitative aspects. For instance, Corpino *et al.* [205] propose a methodology based on the generation of a Quality Function Deployment (QFD) matrix. This QFD matrix evaluates qualitative properties of the system (e.g. sustainability, flexibility and interoperability) as perceived by the stakeholders, with the aim of reducing the gap among the system characteristics and the user needs. However, in the current example of application, the scores are assumed. In particular, each one of the five operating modes of the complete hybrid system might range from 0 to 20. If the operating mode is removed, the relative score is set to 0. Alternatively, this score is set to 15 in case of non-autonomous system, otherwise to 20.

4.5.4 Preliminary design of the alternative system architectures

In this step, the previously identified architectures are preliminarily designed. Therefore, for each architecture a functional model is firstly realized by means of a MBSE approach. Then, the components defined in the Design Synthesis phase of the Harmony process are sized.

The functional model of "Hyb_Sys" has already been described in Section 4.2. Then, a preliminary design of the overall aircraft is performed, analogously to the case study reported in subsection 4.3.3. Focusing on the hybrid system, some results concerning the system components are collected in Table 40.

Table 40: Subset of the preliminary design results of components of "Hyb_Sys" architecture.

Component	Features
Propeller	Max traction: 5000 N; Mass: 25 kg
ICE	Max power: 300 kW; Mass: 200 kg
Battery	Energy: 4000 Wh; Mass: 90 kg
Cooling system	Mass: 55 kg
Electric generator	Max power: 75 kW; Mass: 50 kg

The second solution – "Hyb_No_RAT" – slightly differs from the complete hybrid system case. Minor differences concern only the functional design, as the UC "RecoverEnergy" is completely removed, cancelling all the related functions. This doesn't affect the performance model, which results unchanged. However, the number of functions allocated to the software of the system controller is reduced, entailing a slight decrement of the software cost, as will be shown in the following step of the case study.

The "Hyb_No_Taxi" architecture is represented by the same functional model of the solution characterized by the entire set of capabilities, despite the removal of the operating mode relative to the electric taxi. As an example, it can be clearly seen from the Activity Diagram shown in Figure 35 that even with the removal of the operating mode "GreenTaxi" the functions relative to the storing and the distribution of the electric energy are still present. However, differences are present in the physical model. In fact, the removal of the electric taxi capability entails the reduction of the electric energy storage system. On the other side, a nearly 4 tons airplanes taxing for 2 km at 8.5 m/s would require about 1-1.5 kg of fuel. Thus, performing the OAD of this second architecture entails a new battery mass of 60 kg, sized for the take-off and emergency phases, while the masses of the other components result slightly varied, despite the “snowball” effect.

Concerning the solution “Hyb_No_Aut”, many functions allocated to the controller are deleted. In a non-autonomous system, these functions may be performed by a ground operator instead by the software. The reduction of the autonomy level doesn’t affect the sizing of the hardware components, but reduces the dimensions of the software of the system controller.

In the last alternative solution, both the functional and the performance models are modified with respect to the complete hybrid system case, as the removal of the “GreenTaxi” capability entails a downsize of the electric energy storage, while the elimination of the autonomy functions brings to a reduction of the software size.

4.5.5 Cost estimation of the alternative system architectures

The results obtained from the design of the alternative solutions are employed for the parametric cost estimation of the various components installed in each system architecture. It is worth noting that both the development and the production costs of a single propulsion system depend on the total number of the products planned to be realized. The development costs might be divided per the total number of the systems. Higher is the production volume, lower is the development cost per each product. The same trend applies for the production cost, because of the so-called “learning curve” effect [70]. In other words, the manufactures acquire experience on the aircraft production, making cheaper every following produced airplane. In the present case study, a fleet composed by 35 unmanned hybrid aircraft is assumed.

Like the case of the traditional system architecture, the cost of every component is evaluated by means of the CERs coded within PRICE TruePlanning®, both for the hardware and the software.

In particular, the software cost estimation is typically performed by the number of Source Lines of Code (SLOC). However, this parameter is unknown until more advanced phases of the design process. Therefore, other methods have been derived to attempt a first estimation of the software size earlier in the design. The COSMIC Function Point (CFP) [206] represents a number assessing the size of a certain software based on the number of functionalities implemented. This value can be evaluated from a functional model of the software. The number of CFPs expresses the quantity of information exchanges (or data movement) between the software and the other components of the system and between the software and the external actors. The COSMIC methodology defines four kinds of data movements:

- Entry: data movement directed from an external actor to the software;
- Exit: data movement directed from the software to an external actor;
- Write: data movement directed from the software to a component of the system;
- Read: data movement directed from a component of the system to the software.

Each data movement corresponds to a single CFP. However, in the context of the present doctoral research, a fifth type of CFP is introduced. Other than the previously defined categories of data movements, the count of COSMIC points should include also the operations done by the software, though any information is not exchanged with other components or actors. A processing action indeed might require several lines of code, hence increasing the size of the software. In the current dissertation, this further kind of operation is named "Internal Computation".

The present doctoral thesis proposes an original method for the determination of the number of CFPs based on the functional model of the system under development. The number of data movements and operations made by the software might be directly extracted from one of the SysML diagrams: the White-Box Sequence Diagram. An example of this diagram relative to the UC "Provide propulsive power" is depicted in Figure 61, result of the functional design of a HEPS (Section 4.2).

It is therefore possible to count 5 CFPs from the Sequence Diagram of Figure 61:

- 1 Entry ("reqReceiveThrottleCmd");
- 3 Write ("reqGenerateMechanicalPower1", "reqGenerateMechanical-Power2", and "reqTransferMechanicalPower");
- 2 Internal Computation ("ReceiveThrottleCmd" and "DefineOperating-Mode").

Considering all the four UCs of the "Hyb_Sys" solution, a total number of 45 CFPs is determined. This value, together with the software language, entails the software cost estimation by means of a parametric estimation model. Assuming that

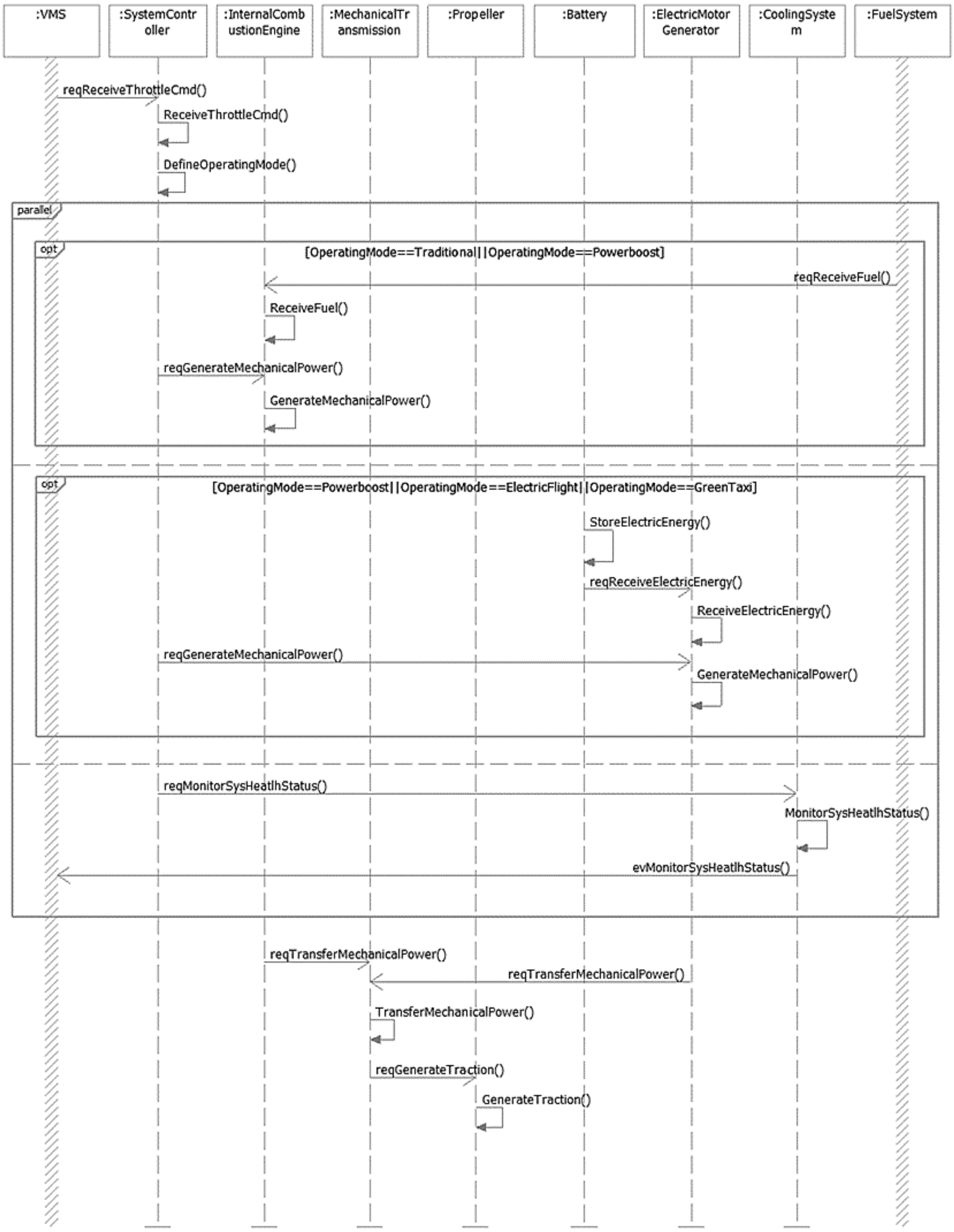


Figure 61: White-Box Sequence Diagram of the UC "Provide propulsive power".

the software is coded in Java language, a total development cost of k€ 1330 results, which corresponds to k€ 38 per system. Concerning the "Hyb_No_RAT" solution, the number of CFPs is slightly decreased in comparison with the complete hybrid system, passing from 45 to 43, hence entailing a software cost of k€ 36.5 per airplane. In particular, the software function of setting the propeller pitch for the maximum efficiency for the energy recovery is removed. The reduction of the system autonomy level entails a more substantial software downsize. The number of CFPs for the "Hyb_No_Aut" and "Hyb_No_Taxi_No_Aut" solutions is 26, which corresponds to a software costing k€ 22 per propulsion system.

The development and production cost – considering both hardware and software components – is estimated for all the alternative solutions. Moreover, the "value" as expressed by eq. 38 is calculated for each architecture, including the traditional one. All these results are reported in Table 41.

Table 41: Cost and "value" of the alternative solutions of propulsion system.

Architecture	**Cost [k€]**	**Value ($x10^4$)**
Trad_Sys	300	0,66
Hyb_Sys	435,5	2,29
Hyb_No_RAT	434	1,84
Hyb_No_Taxi	*395,5*	*2,02*
Hyb_No_Aut	419,5	1,78
Hyb_No_Taxi_No_Aut	*379,5*	*1,97*

4.5.6 Selection of the optimal system architecture

Obviously excluding the traditional architecture, it is worth noting from the results of Table 41 that two alternative solutions are compliant with the target cost requirement: "Hyb_No_Taxi" and "Hyb_No_Taxi_No_Aut". However, although the last solution is the cheapest one, the baseline to be selected should be the hybrid architecture without the capability of performing the green taxi. This configuration is in fact characterized by the higher "value" and then is the solution more appreciated by the customer.

Chapter 5

Conclusions

The research activities described in the current dissertation deal with the following top objective: to develop a successful aeronautical product. This means *in primis* that the new designed aircraft shall be worth more than competitor airplanes belonging to the same market segment. In other words, higher performance and lower operating costs – e.g. due to fuel consumption – shall characterize the new designed product. Retrofits of existing aircraft have been designed to achieve these goals. For instance, the new Airbus A320 neo adopts new and more efficient turbofan engines, entailing reduction of fuel burnt, operating costs and noise/polluting emissions. The on-board systems design discipline can contribute to reduce the aircraft operating costs as well. In this regards, some case studies described in the thesis are referred to the “electrification” of the aircraft on-board systems, with the aim of optimizing the secondary power generation and maximizing the engines efficiency, hence entailing fuel savings. New algorithms for the preliminary design of innovative subsystem architectures have been presented. It has also been demonstrated by means of simpler and “higher-fidelity” design frameworks that unconventional configurations might entail important benefits in terms of fuel reduction. This is especially evident in case of development of “bleedless” architectures, but also solutions characterized by a gross mass reduction due to the removal of the hydraulic power system involve more competitive aircraft. These outcomes are in accordance with journal papers and reports found in literature. Several on-board system architectures have been identified and preliminarily assessed in the thesis. Most of them are at the moment just concepts. However, other configurations can be already considered as state-of-the-art, as the “bleedless” architecture peculiar of the Boeing 787 or the more electric FCS of the Airbus A380. Thus, there is no harm in supposing that future on-board system architectures will be all characterized by innovative elements, as higher electric voltages or adoption of more efficient equipment. Concerning the objective of designing a “better” aircraft, meant as a product characterized by higher performance, an optimization method based on the Fuzzy Logic has been proposed. This methodology aims at negotiating requirements and relaxing constraints – in accordance with the customer and all the involved stakeholders – to improve other aircraft specifications. For instance, in the test case proposed in the thesis, improvements have been obtained in terms of fuel mass reduction and safety level increase.

Another aspect required to new aircraft is the reduction of environmental impact. As the automotive field, also in aviation greener solutions are currently investigated. Airplanes powered by only electric propulsion systems are already flying, though are characterized by low range and endurance. The weak point of these innovative concepts is indeed represented by the current technology of electric energy storage systems, which are affected by disadvantageous energy-to-weight ratios. Hybrid-electric powered aircraft seem to be at the moment one of the most promising solutions able to partly gain the benefits of an electric propulsion system without losing aircraft performance. In particular, hybrid-electric propulsion aircraft move towards a more environmental friendly solution. To be fair, it must be said that the quantity of pollution produced by aircraft – and in particular by general aviation airplanes – represents a small percentage of the global quantity of polluting emissions. In 1999 the level of CO_2 emissions due to civil transport aviation has been estimated around 2% [207]. The percentage of emission quantity might rise up to 4% considering also NOx, particulates and other emissions [208]. However, several studies are devoted to the exploitation of the hybrid-electric solution on larger airplanes, as the E-Fan X demonstrator, a retrofit version of the regional aircraft BAe 146, realized in collaboration by Airbus, Rolls-Royce and Siemens [209]. As a hybrid-electric aircraft design methodology is missing in literature, an original one has been proposed in the current dissertation. This methodology aims at preliminarily sizing a parallel hybrid-electric propulsion system, assessing the impacts and possible benefits of this technology on the entire aircraft level. By means of test cases, it has been proven that hybrid-electric powered aircraft might lead to greener and safer solutions.

The success of new aircraft is also enabled by the development process, other than the product itself. Aeronautical companies have to deal with competitors, and additional high level requirements shall be identified at the beginning of the design phase. One of these requirements is the maximum development and production cost. A DTC approach has been followed in the dissertation to perform a trade-off analysis among different subsystem architectures, selecting the one compliant with the cost and all the other requirements. This approach shall be adopted since the very beginning of the design process, as in this phase the higher majority of costs is allocated. However, costs monitoring shall be performed all along the entire design process, making modifications when discrepancies with the target and the effective costs are pointed out.

The TTM is another strict design requirement. Delays in the product delivery would indeed entail penalties and low sales, therefore involving low profits. The research studies performed within the context of the current doctoral activity have been focused on the definition and set up of a so-called 3rd Generation design framework. It consists of an innovative multidisciplinary, collaborative and distributed aircraft design environment, which is conceived to automate various design processes, entailing a strong reduction of the design phases. In particular, this thesis is focused on the integration of the subsystems design discipline within the framework, with the aim of assessing the impacts of this discipline on the other disciplines and vice-versa. Effects on fuel consumption due to the various kinds of secondary power extraction have been preliminarily quantified, identifying the bleed air off-take as the most detrimental one from an engine efficiency – and thus from a fuel consumption – perspective. Additional design techniques might be employed in the setup of this design framework to accelerate the development process. For instance, RSMs can be constructed and utilized to preliminary assess the response of disciplinary models that would require hours or even days to compute a solution. The response surface of an entire aircraft conceptual design model has been realized within the context of the present doctoral dissertation. It has shown that this RSM entails the evaluation of the main aircraft specifications – as design masses – returning solutions affected by a little percentage of error.

It should be noted that the product total development and production cost and the delivery time might be reduced also modifying or introducing new manufacturing processes, as the “friction stir welding”. However, the current dissertation has been focused mainly on the conceptual and preliminary design of the product itself, neglecting the production aspects, which shall be indeed accounted during the development of the new aircraft.

The designed product shall indeed comply with all the needs and the expectations required by all the involved stakeholders. Moreover, functionalities of the new aircraft that don't bring any added value from the customer perspective shall be avoided. The wrong or incomplete elicitation of all the product requirements and the scarce involvement of all the stakeholders represent the most relevant causes of typical project failure. Therefore, a SE based methodology has been adopted to derive and develop from a set of high level requirements all the system functionalities effectively requested by the stakeholders. This approach would minimize the number of design errors, hence preventing re-designs and modifications and consequently avoiding delays and additional development costs. The proposed methodology allows the definition of the MDO problem, identifying all the disciplines that should be available to reach the design objectives, connecting them within a development framework and defining a design and optimization strategy.

Various are the contributions achieved within the research activities described in the dissertation. Nevertheless, future work might be carried out to improve and exploit the achievements and to overcome the limitations of the doctoral dissertation. In particular, original algorithms for the preliminary sizing of parallel HEPS have been developed. Other architectures of hybrid propulsion systems might be considered in aeronautics, as the serial hybrid one. However the design methodology here proposed might represent a starting point for future research. In addition, case studies shall be conducted on larger aircraft with different propulsion systems, for instance small regional turboprop airplanes. Existing algorithms have also been enhanced for the preliminary sizing of innovative (i.e. More and All Electric) on-board systems. Again, these algorithms are restricted to civil passenger transport aircraft. Considerations for other classes of airplanes – such as military fighters – must be assessed. Moreover, the methodologies for the preliminary design of innovative configurations are affected by the scarcity and unavailability of data and information concerning these kinds of architectures. However, the proposed algorithms entail a first evaluation, although they can be improved once additional data is made available. Furthermore, sensitivity analyses shall be performed to investigate the impact of design uncertainties affecting the subsystem and aircraft results. Additionally, only evaluations in terms of subsystem masses and power off-takes have been made. However, additional results shall be derived to completely design the on-board systems, as volumes, installation constraints, RAMS evaluations, costs. Consequently, all the impacts of subsystems design at aircraft level can be properly investigated and quantified, eventually determining the best on-board system architecture.

The activities performed within the context of the present doctoral dissertation have also contributed to the realization of an innovative 3rd Generation MDO framework. In the present work the attention is mainly focused on the integration of the subsystems design discipline. More details are present in the journal papers and conference proceedings published by the Consortium of the AGILE project (a complete list of all these publications is reported in the AGILE project website). In addition, MBSE techniques have been studied and employed for the complete determination of all the system capabilities actually required by the stakeholders. It has also been proposed a Fuzzy Logic based method for the aircraft multi-objective optimization and for the definition of the best solution from the customer perspective, by negotiating system capabilities and high level requirements.

Finally, additional achievements have been derived by means of the employment of the proposed design techniques and methodology. First, the optimal hybrid configuration has been defined, meant as the safest solution or the most fuel-efficient or the most cost-effective one. Then, the most fuel-efficient and the lightest on-board system architectures for a 90-passenger transport regional jet have been assessed. All these outcomes prove the effectiveness and the usefulness of the proposed design techniques.

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